# CHAPTER

# 57

# WINGS

# CHAPTER 57
# WINGS



A = Added, R = Revised, O = Overflow, D = Deleted

## CHAPTER 57
## WINGS

| Subject/Page | Date | Subject/Page | Date | Subject/Page | Date |
|---|---|---|---|---|---|
| 57-10-03 REPAIR 6 (cont) | | 57-10-10 ALLOWABLE DAMAGE 2 | | 57-10-10 REPAIR 2 (cont) | |
| 202 | Jul 10/2005 | 101 | Jul 10/2007 | R 218 | Jul 10/2009 |
| 203 | Nov 01/2003 | 102 | Mar 10/2004 | O 219 | Jul 10/2009 |
| 204 | Nov 01/2003 | R 103 | Jul 10/2009 | O 220 | Jul 10/2009 |
| 205 | Nov 01/2003 | R 104 | Jul 10/2009 | R 221 | Jul 10/2009 |
| 206 | Nov 01/2003 | R 105 | Jul 10/2009 | O 222 | Jul 10/2009 |
| 207 | Nov 01/2003 | R 106 | Jul 10/2009 | O 223 | Jul 10/2009 |
| 208 | Jul 10/2005 | R 107 | Jul 10/2009 | O 224 | Jul 10/2009 |
| 57-10-03 REPAIR 7 | | R 108 | Jul 10/2009 | O 225 | Jul 10/2009 |
| 201 | Nov 01/2003 | O 109 | Jul 10/2009 | O 226 | Jul 10/2009 |
| 202 | Jul 10/2005 | O 110 | Jul 10/2009 | R 227 | Jul 10/2009 |
| 203 | Nov 01/2003 | O 111 | Jul 10/2009 | O 228 | Jul 10/2009 |
| 204 | Nov 01/2003 | O 112 | Jul 10/2009 | O 229 | Jul 10/2009 |
| 205 | Nov 01/2003 | O 113 | Jul 10/2009 | R 230 | Jul 10/2009 |
| 206 | Nov 01/2003 | O 114 | Jul 10/2009 | O 231 | Jul 10/2009 |
| 207 | Jul 10/2005 | O 115 | Jul 10/2009 | O 232 | Jul 10/2009 |
| 208 | BLANK | O 116 | Jul 10/2009 | O 233 | Jul 10/2009 |
| 57-10-10 IDENTIFICATION 1 | | A 117 | Jul 10/2009 | O 234 | Jul 10/2009 |
| 1 | Mar 10/2004 | A 118 | BLANK | O 235 | Jul 10/2009 |
| 2 | Mar 10/2004 | 57-10-10 REPAIR 1 | | R 236 | Jul 10/2009 |
| 3 | Nov 10/2006 | 201 | Nov 10/2006 | O 237 | Jul 10/2009 |
| 4 | BLANK | 202 | BLANK | R 238 | Jul 10/2009 |
| 57-10-10 IDENTIFICATION 2 | | 57-10-10 REPAIR 2 | | O 239 | Jul 10/2009 |
| 1 | Mar 10/2004 | R 201 | Jul 10/2009 | O 240 | Jul 10/2009 |
| 2 | Jul 10/2005 | R 202 | Jul 10/2009 | O 241 | Jul 10/2009 |
| 3 | Jul 10/2005 | 203 | Jul 10/2007 | R 242 | Jul 10/2009 |
| 4 | Nov 10/2006 | 204 | Jul 10/2007 | R 243 | Jul 10/2009 |
| 5 | Nov 10/2006 | R 205 | Jul 10/2009 | R 244 | Jul 10/2009 |
| 6 | BLANK | R 206 | Jul 10/2009 | O 245 | Jul 10/2009 |
| 57-10-10 ALLOWABLE DAMAGE 1 | | O 207 | Jul 10/2009 | R 246 | Jul 10/2009 |
| 101 | Nov 01/2003 | O 208 | Jul 10/2009 | O 247 | Jul 10/2009 |
| 102 | Nov 01/2003 | R 209 | Jul 10/2009 | O 248 | Jul 10/2009 |
| 103 | Jul 10/2004 | R 210 | Jul 10/2009 | R 249 | Jul 10/2009 |
| 104 | Jul 10/2004 | R 211 | Jul 10/2009 | O 250 | Jul 10/2009 |
| 105 | Jul 10/2004 | O 212 | Jul 10/2009 | O 251 | Jul 10/2009 |
| 106 | Nov 01/2003 | O 213 | Jul 10/2009 | O 252 | Jul 10/2009 |
| 107 | Nov 01/2003 | O 214 | Jul 10/2009 | O 253 | Jul 10/2009 |
| 108 | Nov 01/2003 | R 215 | Jul 10/2009 | O 254 | Jul 10/2009 |
| 109 | Nov 01/2003 | O 216 | Jul 10/2009 | O 255 | Jul 10/2009 |
| 110 | Nov 01/2003 | O 217 | Jul 10/2009 | R 256 | Jul 10/2009 |

A = Added, R = Revised, O = Overflow, D = Deleted

# CHAPTER 57
# WINGS



A = Added, R = Revised, O = Overflow, D = Deleted

# CHAPTER 57
# WINGS

| Subject/Page | Date |
|---|---|
| 57-20-03 REPAIR 3 | |
| 201 | Nov 01/2003 |
| 202 | Nov 01/2003 |
| 203 | Nov 01/2003 |
| 204 | Nov 01/2003 |
| 205 | Nov 01/2003 |
| 206 | Nov 01/2003 |
| 57-20-03 REPAIR 4 | |
| 201 | Nov 01/2003 |
| 202 | Nov 01/2003 |
| 203 | Nov 01/2003 |
| 204 | Nov 01/2003 |
| 205 | Nov 01/2003 |
| 206 | Nov 01/2003 |
| 57-20-03 REPAIR 5 | |
| 201 | Nov 01/2003 |
| 202 | Nov 01/2003 |
| 203 | Nov 01/2003 |
| 204 | Nov 01/2003 |
| 205 | Nov 01/2003 |
| 206 | Nov 01/2003 |
| 207 | Nov 01/2003 |
| 208 | Nov 01/2003 |
| 57-20-09 IDENTIFICATION 1 | |
| 1 | Jul 10/2004 |
| 2 | Mar 10/2007 |
| 3 | Mar 10/2007 |
| 4 | Mar 10/2004 |
| 5 | Mar 10/2004 |
| 6 | Nov 01/2003 |
| 7 | Nov 01/2003 |
| 8 | Mar 10/2004 |
| 9 | Mar 10/2004 |
| 10 | Jul 10/2004 |
| 57-20-09 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Jul 10/2004 |
| 104 | Jul 10/2004 |
| 105 | Jul 10/2004 |
| 57-20-09 ALLOWABLE DAMAGE 1 (cont) | |
| 106 | Nov 01/2003 |
| 107 | Nov 01/2003 |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 110 | BLANK |
| 57-20-09 REPAIR 1 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-20-10 IDENTIFICATION 1 | |
| 1 | Mar 10/2007 |
| 2 | Mar 10/2007 |
| 3 | Mar 10/2007 |
| 4 | Nov 10/2006 |
| 5 | Nov 10/2006 |
| 6 | BLANK |
| 57-20-10 IDENTIFICATION 2 | |
| 1 | Mar 10/2007 |
| 2 | Mar 10/2007 |
| 3 | Mar 10/2007 |
| 4 | Mar 10/2004 |
| 5 | Mar 10/2004 |
| 6 | BLANK |
| 57-20-10 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Jul 10/2007 |
| 106 | Nov 01/2003 |
| 107 | Jul 10/2007 |
| 108 | Jul 10/2007 |
| 109 | Jul 10/2007 |
| 110 | Jul 10/2007 |
| 111 | Jul 10/2007 |
| 112 | BLANK |
| 57-20-10 ALLOWABLE DAMAGE 2 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 57-20-10 ALLOWABLE DAMAGE 2 (cont) | |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Jul 10/2007 |
| 106 | Nov 01/2003 |
| 107 | Jul 10/2007 |
| 108 | Jul 10/2007 |
| 109 | Jul 10/2007 |
| 110 | Jul 10/2007 |
| 111 | Jul 10/2007 |
| 112 | BLANK |
| 57-20-10 REPAIR 1 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-20-90 IDENTIFICATION 1 | |
| 1 | Jul 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Nov 01/2003 |
| 4 | Nov 01/2003 |
| 57-20-90 IDENTIFICATION 2 | |
| 1 | Nov 10/2006 |
| 2 | Mar 10/2004 |
| 3 | Nov 01/2003 |
| 4 | Nov 01/2003 |
| 57-20-90 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 10/2007 |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Nov 01/2003 |
| 106 | Nov 01/2003 |
| 107 | Nov 01/2003 |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 110 | Nov 01/2003 |
| 111 | Nov 01/2003 |
| 112 | Nov 01/2003 |
| 113 | Nov 01/2003 |
| 114 | Nov 01/2003 |

A = Added, R = Revised, O = Overflow, D = Deleted

## CHAPTER 57
## WINGS



A = Added, R = Revised, O = Overflow, D = Deleted

| Subject/Page | Date |
|---|---|
| 57-30-02 ALLOWABLE DAMAGE 2 (cont) | |
| 106 | Nov 10/2003 |
| 107 | Nov 10/2003 |
| 108 | BLANK |
| 57-30-02 REPAIR 1 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-30-02 REPAIR 2 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-41-01 IDENTIFICATION 1 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Mar 10/2004 |
| 5 | Nov 01/2003 |
| 6 | BLANK |
| 57-41-01 IDENTIFICATION 2 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Mar 10/2004 |
| 5 | Mar 10/2004 |
| 6 | Mar 10/2004 |
| 57-41-01 IDENTIFICATION 3 | |
| 1 | Nov 10/2007 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Mar 10/2004 |
| 5 | Mar 10/2004 |
| 6 | Mar 10/2004 |
| 7 | Mar 10/2004 |
| 8 | Mar 10/2004 |
| 9 | Mar 10/2004 |
| 10 | Mar 10/2004 |
| 11 | Mar 10/2004 |
| 12 | Mar 10/2004 |
| 13 | Mar 10/2004 |
| 14 | Mar 10/2004 |
| 57-41-01 IDENTIFICATION 3 (cont) | |
| 15 | Mar 10/2004 |
| 16 | Nov 01/2003 |
| 17 | Mar 10/2004 |
| 18 | Mar 10/2004 |
| 19 | Nov 01/2003 |
| 20 | Mar 10/2004 |
| 21 | Mar 10/2004 |
| 22 | Nov 01/2003 |
| 23 | Mar 10/2004 |
| 24 | Mar 10/2004 |
| 25 | Nov 10/2006 |
| 26 | Nov 01/2003 |
| 57-41-01 IDENTIFICATION 4 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Nov 10/2006 |
| 4 | Nov 10/2006 |
| 5 | Mar 10/2004 |
| 6 | Nov 10/2006 |
| 7 | Mar 10/2004 |
| 8 | Nov 10/2006 |
| 9 | Nov 10/2006 |
| 10 | Nov 10/2006 |
| 11 | Nov 10/2006 |
| 12 | Mar 10/2004 |
| 13 | Nov 10/2006 |
| 14 | Mar 10/2004 |
| 15 | Nov 10/2006 |
| 16 | Nov 10/2006 |
| 17 | Nov 10/2006 |
| 18 | Nov 10/2006 |
| 19 | Nov 10/2006 |
| 20 | Nov 10/2006 |
| 21 | Nov 10/2006 |
| 22 | Nov 10/2006 |
| 23 | Nov 10/2006 |
| 24 | Nov 10/2006 |
| 25 | Nov 10/2006 |
| 26 | Nov 10/2006 |
| 57-41-01 IDENTIFICATION 4 (cont) | |
| 27 | Nov 10/2006 |
| 28 | Nov 10/2006 |
| 29 | Nov 10/2006 |
| 30 | Nov 10/2006 |
| 31 | Nov 10/2006 |
| 32 | Nov 10/2006 |
| 33 | Nov 10/2006 |
| 34 | Nov 10/2006 |
| 35 | Nov 10/2006 |
| 36 | Nov 10/2006 |
| 37 | Nov 10/2006 |
| 38 | Nov 10/2006 |
| 39 | Nov 10/2006 |
| 40 | Nov 10/2006 |
| 41 | Nov 10/2006 |
| 42 | Nov 10/2006 |
| 57-41-01 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Jul 10/2005 |
| 104 | Nov 10/2007 |
| 105 | Nov 10/2006 |
| 106 | Jul 10/2004 |
| 107 | Nov 01/2003 |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 110 | Nov 01/2003 |
| 111 | Nov 01/2003 |
| 112 | Nov 10/2007 |
| 113 | Nov 10/2006 |
| 114 | Nov 10/2006 |
| 115 | Nov 10/2006 |
| 116 | Nov 10/2006 |
| 117 | Nov 10/2006 |
| 118 | Nov 01/2003 |
| 119 | Nov 10/2006 |
| 120 | Nov 01/2003 |
| 121 | Nov 01/2003 |
| 122 | Nov 01/2003 |

A = Added, R = Revised, O = Overflow, D = Deleted

CHAPTER 57
WINGS



A = Added, R = Revised, O = Overflow, D = Deleted

CHAPTER 57
WINGS

| Subject/Page | Date | Subject/Page | Date | Subject/Page | Date |
|---|---|---|---|---|---|
| 57-42-01 ALLOWABLE DAMAGE 1 (cont) | | 57-42-02 IDENTIFICATION 1 (cont) | | 57-43-02 ALLOWABLE DAMAGE 1 (cont) | |
| 111 | Nov 01/2003 | 5 | Nov 10/2006 | 106 | Nov 01/2003 |
| 112 | Nov 01/2003 | 6 | Nov 10/2006 | 107 | Nov 01/2003 |
| 113 | Nov 01/2003 | 57-42-02 ALLOWABLE DAMAGE 1 | | 108 | BLANK |
| 114 | Nov 01/2003 | 101 | Nov 01/2003 | 57-43-02 REPAIR 1 | |
| 115 | Nov 01/2003 | 102 | Nov 01/2003 | 201 | Nov 10/2007 |
| 116 | BLANK | 103 | Nov 01/2003 | 202 | Nov 10/2007 |
| 57-42-01 REPAIR 1 | | 104 | Nov 01/2003 | 203 | Nov 10/2007 |
| 201 | Mar 10/2004 | 105 | Nov 10/2007 | 204 | Nov 10/2006 |
| 202 | Jul 10/2005 | 106 | Nov 10/2007 | 57-43-02 REPAIR 2 | |
| 203 | Nov 01/2003 | 107 | Jul 10/2004 | 201 | Nov 10/2007 |
| 204 | Nov 01/2003 | 108 | Nov 01/2003 | 202 | Nov 10/2007 |
| 205 | Nov 01/2003 | 109 | Nov 01/2003 | 203 | Nov 10/2007 |
| 206 | Mar 10/2004 | 110 | Nov 01/2003 | 204 | Nov 10/2006 |
| 57-42-01 REPAIR 2 | | 111 | Nov 01/2003 | 205 | Nov 01/2003 |
| 201 | Nov 01/2003 | 112 | Nov 01/2003 | 206 | BLANK |
| 202 | Jul 10/2005 | 113 | Nov 01/2003 | 57-43-90 IDENTIFICATION 1 | |
| 203 | Nov 01/2003 | 114 | Nov 01/2003 | 1 | Jul 10/2004 |
| 204 | Nov 01/2003 | 57-42-02 REPAIR 1 | | 2 | Mar 10/2004 |
| 205 | Nov 01/2003 | 201 | Nov 10/2006 | 3 | Nov 10/2006 |
| 206 | BLANK | 202 | Nov 10/2007 | 4 | Mar 10/2004 |
| 57-42-01 REPAIR 3 | | 203 | Nov 10/2007 | 5 | Nov 10/2006 |
| 201 | Nov 10/2006 | 204 | Nov 01/2003 | 6 | BLANK |
| 202 | BLANK | 205 | Nov 01/2003 | 57-43-90 ALLOWABLE DAMAGE 1 | |
| 57-42-01 REPAIR 4 | | 206 | Nov 01/2003 | 101 | Nov 01/2003 |
| 201 | Nov 01/2003 | 207 | Nov 01/2003 | 102 | Nov 01/2003 |
| 202 | Nov 01/2003 | 208 | BLANK | 103 | Nov 01/2003 |
| 57-42-01 REPAIR 5 | | 57-43-02 IDENTIFICATION 1 | | 104 | Nov 01/2003 |
| 201 | Nov 10/2004 | 1 | Mar 10/2004 | 105 | Nov 01/2003 |
| 202 | Nov 10/2004 | 2 | Mar 10/2004 | 106 | Nov 01/2003 |
| 203 | Nov 10/2004 | 3 | Mar 10/2004 | 107 | Nov 01/2003 |
| 204 | Nov 10/2004 | 4 | Mar 10/2004 | 108 | BLANK |
| 205 | Mar 10/2006 | 5 | Mar 10/2004 | 57-43-90 REPAIR 1 | |
| 206 | BLANK | 6 | Nov 01/2003 | 201 | Nov 10/2006 |
| 57-42-02 IDENTIFICATION 1 | | 57-43-02 ALLOWABLE DAMAGE 1 | | 202 | Nov 10/2006 |
| 1 | Mar 10/2004 | 101 | Nov 01/2003 | 57-51-01 IDENTIFICATION 1 | |
| 2 | Mar 10/2004 | 102 | Nov 01/2003 | 1 | Nov 10/2006 |
| 3 | Nov 10/2006 | 103 | Nov 01/2003 | 2 | Nov 10/2006 |
| 4 | Nov 10/2006 | 104 | Jul 10/2004 | 3 | Nov 10/2006 |
| | | 105 | Nov 01/2003 | | |

A = Added, R = Revised, O = Overflow, D = Deleted

**CHAPTER 57**
**WINGS**

| Subject/Page | Date | Subject/Page | Date | Subject/Page | Date |
|---|---|---|---|---|---|
| 57-51-01 IDENTIFICATION 1 (cont) | | 57-51-02 IDENTIFICATION 1 | | 57-51-02 REPAIR 1 (cont) | |
| 4 | Jul 10/2007 | 1 | Nov 10/2006 | 208 | BLANK |
| 5 | Nov 10/2006 | 2 | Nov 10/2006 | 57-51-14 IDENTIFICATION 1 | |
| 6 | Nov 10/2006 | 3 | Nov 10/2006 | 1 | Nov 10/2006 |
| 7 | Nov 10/2006 | 4 | Nov 10/2006 | 2 | Mar 10/2004 |
| 8 | Nov 10/2006 | 5 | Nov 10/2006 | 3 | Nov 10/2006 |
| 9 | Jul 10/2007 | 6 | Nov 10/2006 | 4 | Nov 10/2006 |
| 10 | Nov 10/2006 | 7 | Nov 10/2006 | 57-51-14 ALLOWABLE DAMAGE 1 | |
| 11 | Nov 10/2006 | 8 | Nov 10/2006 | 101 | Nov 01/2003 |
| 12 | Nov 10/2006 | 9 | Nov 10/2006 | 102 | Nov 10/2006 |
| 13 | Nov 10/2006 | 10 | Nov 10/2006 | 103 | Nov 01/2003 |
| 14 | Nov 10/2006 | 11 | Nov 10/2006 | 104 | Nov 01/2003 |
| 57-51-01 ALLOWABLE DAMAGE 1 | | 12 | Nov 10/2006 | 105 | Nov 10/2006 |
| 101 | Jul 10/2005 | 13 | Nov 10/2006 | 106 | Nov 10/2006 |
| 102 | Nov 10/2007 | 14 | BLANK | 107 | Nov 01/2003 |
| 103 | Nov 10/2004 | 57-51-02 ALLOWABLE DAMAGE 1 | | 108 | Nov 01/2003 |
| 104 | Nov 01/2003 | 101 | Nov 10/2007 | 109 | Nov 01/2003 |
| 105 | Nov 01/2003 | 102 | Nov 10/2007 | 110 | BLANK |
| 106 | Nov 01/2003 | 103 | Nov 10/2007 | 57-51-90 IDENTIFICATION 1 | |
| 107 | Nov 10/2007 | 104 | Nov 10/2007 | 1 | Nov 10/2006 |
| 108 | Jul 10/2004 | 105 | Nov 10/2007 | 2 | Nov 10/2006 |
| 109 | Jul 10/2004 | 106 | Nov 10/2007 | 3 | Nov 10/2006 |
| 110 | Jul 10/2004 | 107 | Nov 10/2007 | 4 | Nov 10/2006 |
| 111 | Jul 10/2004 | 108 | Mar 10/2008 | 5 | Nov 10/2006 |
| 112 | Jul 10/2004 | 109 | Mar 10/2008 | 6 | BLANK |
| 113 | Jul 10/2004 | 110 | Nov 10/2007 | 57-51-90 ALLOWABLE DAMAGE 1 | |
| 114 | Nov 01/2003 | 111 | Nov 10/2007 | 101 | Nov 10/2007 |
| 115 | Nov 01/2003 | 112 | Nov 10/2007 | 102 | Nov 10/2007 |
| 116 | Nov 01/2003 | 113 | Nov 10/2007 | 103 | Nov 10/2007 |
| 117 | Nov 01/2003 | 114 | Nov 10/2007 | 104 | Nov 10/2007 |
| 118 | Nov 01/2003 | 115 | Nov 10/2007 | 105 | Nov 10/2007 |
| 57-51-01 REPAIR 1 | | 116 | Nov 10/2007 | 106 | Nov 10/2007 |
| 201 | Nov 01/2003 | 57-51-02 REPAIR 1 | | 57-53-01 IDENTIFICATION 1 | |
| 202 | Nov 01/2003 | 201 | Mar 10/2005 | 1 | Mar 10/2004 |
| 203 | Nov 01/2003 | 202 | Mar 10/2005 | 2 | Mar 10/2004 |
| 204 | Jul 10/2005 | 203 | Nov 10/2008 | 3 | Jul 10/2005 |
| 205 | Nov 01/2003 | 204 | Jul 10/2007 | 4 | Mar 10/2004 |
| 206 | Jul 10/2005 | 205 | Mar 10/2005 | 5 | Mar 10/2004 |
| 207 | Jul 10/2004 | 206 | Jul 10/2007 | 6 | Mar 10/2004 |
| 208 | BLANK | 207 | Nov 10/2008 | 7 | Mar 10/2004 |

A = Added, R = Revised, O = Overflow, D = Deleted

## CHAPTER 57
## WINGS

| Subject/Page | Date | Subject/Page | Date | Subject/Page | Date |
|---|---|---|---|---|---|
| 57-53-01 IDENTIFICATION 1 (cont) | | 57-53-01 ALLOWABLE DAMAGE 1 (cont) | | 57-53-01 ALLOWABLE DAMAGE 4 (cont) | |
| 8 | Mar 10/2004 | 109 | Jul 10/2004 | 110 | Mar 10/2004 |
| 9 | Mar 10/2004 | 110 | Jul 10/2004 | 111 | Mar 10/2004 |
| 10 | Mar 10/2004 | 111 | Jul 10/2004 | 112 | Mar 10/2004 |
| 11 | Mar 10/2004 | 112 | BLANK | 57-53-01 ALLOWABLE DAMAGE 5 | |
| 12 | Mar 10/2004 | 57-53-01 ALLOWABLE DAMAGE 2 | | 101 | Nov 10/2005 |
| 13 | Mar 10/2004 | 101 | Jul 10/2005 | 102 | Nov 10/2005 |
| 14 | BLANK | R 102 | Jul 10/2009 | 103 | Nov 10/2005 |
| 57-53-01 IDENTIFICATION 2 | | 103 | Jul 10/2004 | 104 | Nov 10/2005 |
| 1 | Nov 10/2004 | 104 | Jul 10/2004 | 105 | Nov 10/2005 |
| 2 | Nov 10/2004 | 105 | Nov 01/2003 | 106 | Nov 10/2005 |
| R 3 | Jul 10/2009 | 106 | Nov 01/2003 | 107 | Nov 10/2005 |
| 4 | Nov 10/2004 | 107 | Nov 01/2003 | 108 | Nov 10/2005 |
| 57-53-01 IDENTIFICATION 3 | | 108 | Nov 01/2003 | 109 | Nov 10/2005 |
| 1 | Nov 10/2004 | 57-53-01 ALLOWABLE DAMAGE 3 | | 110 | BLANK |
| 2 | Nov 10/2004 | 101 | Nov 01/2003 | 57-53-01 REPAIR 1 | |
| 3 | Nov 10/2004 | 102 | Nov 01/2003 | 201 | Nov 10/2007 |
| 4 | Nov 10/2004 | 103 | Nov 01/2003 | 202 | Nov 01/2003 |
| 5 | Nov 10/2004 | 104 | Nov 10/2007 | 203 | Nov 10/2007 |
| 6 | Nov 10/2004 | 105 | Nov 10/2007 | 204 | Nov 10/2007 |
| 7 | Nov 10/2004 | 106 | Nov 10/2007 | 205 | Nov 01/2003 |
| 8 | Nov 10/2004 | 107 | Nov 10/2007 | 206 | BLANK |
| 9 | Nov 10/2004 | 108 | Nov 10/2007 | 57-53-01 REPAIR 2 | |
| 10 | Nov 10/2004 | 109 | Nov 10/2007 | 201 | Nov 01/2003 |
| 11 | Nov 10/2004 | 110 | Nov 10/2007 | 202 | Nov 01/2003 |
| 12 | BLANK | 111 | Nov 10/2007 | 203 | Nov 10/2007 |
| 57-53-01 IDENTIFICATION 4 | | 112 | Nov 10/2007 | 204 | Nov 10/2007 |
| 1 | Nov 10/2004 | 113 | Nov 10/2007 | 205 | Nov 01/2003 |
| 2 | Nov 10/2004 | 114 | BLANK | 206 | Nov 01/2003 |
| R 3 | Jul 10/2009 | 57-53-01 ALLOWABLE DAMAGE 4 | | 207 | Nov 01/2003 |
| 4 | Nov 10/2004 | 101 | Jul 10/2005 | 208 | BLANK |
| 57-53-01 ALLOWABLE DAMAGE 1 | | R 102 | Jul 10/2009 | 57-53-01 REPAIR 3 | |
| 101 | Nov 01/2003 | 103 | Mar 10/2004 | 201 | Nov 01/2003 |
| 102 | Jul 10/2004 | 104 | Mar 10/2005 | 202 | BLANK |
| 103 | Nov 10/2007 | 105 | Jul 10/2004 | 57-53-01 REPAIR 4 | |
| 104 | Nov 10/2007 | 106 | Jul 10/2004 | 201 | Nov 10/2004 |
| 105 | Jul 10/2004 | 107 | Jul 10/2004 | 202 | BLANK |
| 106 | Jul 10/2004 | 108 | Mar 10/2004 | 57-53-02 IDENTIFICATION 1 | |
| 107 | Jul 10/2004 | 109 | Mar 10/2004 | 1 | Mar 10/2004 |
| 108 | Jul 10/2004 | | | | |

A = Added, R = Revised, O = Overflow, D = Deleted

# CHAPTER 57
# WINGS

| Subject/Page | Date |
|---|---|
| 57-53-02 IDENTIFICATION 1 (cont) | |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Nov 01/2003 |
| 5 | Nov 01/2003 |
| 6 | Nov 01/2003 |
| 7 | Nov 01/2003 |
| 8 | Mar 10/2004 |
| 9 | Nov 01/2003 |
| 10 | BLANK |
| 57-53-02 IDENTIFICATION 2 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Nov 01/2003 |
| 57-53-02 IDENTIFICATION 3 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Nov 01/2003 |
| 5 | Nov 01/2003 |
| 6 | Nov 01/2003 |
| 57-53-02 IDENTIFICATION 4 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Nov 01/2003 |
| 5 | Nov 01/2003 |
| 6 | BLANK |
| 57-53-02 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Mar 10/2009 |
| 103 | Mar 10/2009 |
| 104 | Mar 10/2009 |
| 105 | Mar 10/2009 |
| 106 | Mar 10/2009 |
| 107 | Mar 10/2009 |
| 108 | Mar 10/2009 |
| 109 | Mar 10/2009 |
| 110 | Mar 10/2009 |
| 57-53-02 ALLOWABLE DAMAGE 1 (cont) | |
| 111 | Mar 10/2009 |
| 112 | Mar 10/2009 |
| 57-53-02 ALLOWABLE DAMAGE 2 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Mar 10/2005 |
| 104 | Jul 10/2004 |
| 105 | Nov 01/2003 |
| 106 | Nov 01/2003 |
| 107 | Nov 01/2003 |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 110 | Nov 01/2003 |
| 111 | Nov 01/2003 |
| 112 | Nov 01/2003 |
| 57-53-02 ALLOWABLE DAMAGE 3 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Nov 10/2006 |
| 106 | Jul 10/2004 |
| 107 | Jul 10/2004 |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 110 | Nov 01/2003 |
| 111 | Nov 01/2003 |
| 112 | BLANK |
| 57-53-02 ALLOWABLE DAMAGE 4 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Jul 10/2004 |
| 104 | Jul 10/2004 |
| 105 | Nov 01/2003 |
| 106 | Nov 01/2003 |
| 107 | Nov 01/2003 |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 57-53-02 ALLOWABLE DAMAGE 4 (cont) | |
| 110 | Nov 01/2003 |
| 111 | Nov 01/2003 |
| 112 | BLANK |
| 57-53-02 REPAIR 1 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-53-02 REPAIR 2 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-53-02 REPAIR 3 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-53-02 REPAIR 4 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-53-02 REPAIR 5 | |
| A 201 | Jul 10/2009 |
| A 202 | Jul 10/2009 |
| A 203 | Jul 10/2009 |
| A 204 | Jul 10/2009 |
| A 205 | Jul 10/2009 |
| A 206 | Jul 10/2009 |
| A 207 | Jul 10/2009 |
| A 208 | Jul 10/2009 |
| 57-53-70 IDENTIFICATION 1 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Nov 10/2006 |
| 5 | Mar 10/2004 |
| 6 | Mar 10/2004 |
| 7 | Mar 10/2004 |
| 8 | Mar 10/2004 |
| 9 | Mar 10/2004 |
| 10 | Mar 10/2004 |
| 11 | Mar 10/2004 |
| 12 | Mar 10/2004 |
| 13 | Mar 10/2004 |

A = Added, R = Revised, O = Overflow, D = Deleted

# CHAPTER 57
# WINGS

| Subject/Page | Date | Subject/Page | Date | Subject/Page | Date |
|---|---|---|---|---|---|
| 57-53-70 IDENTIFICATION 1 (cont) | | 57-53-70 IDENTIFICATION 3 (cont) | | 57-53-70 REPAIR 1 | |
| 14 | Mar 10/2004 | 5 | Mar 10/2004 | 201 | Nov 10/2007 |
| 15 | Mar 10/2004 | 6 | Nov 10/2006 | 202 | Jul 10/2004 |
| 16 | Mar 10/2004 | 7 | Mar 10/2004 | 203 | Nov 01/2003 |
| 17 | Mar 10/2004 | 8 | Nov 10/2006 | 204 | Nov 01/2003 |
| 18 | Mar 10/2004 | 9 | Mar 10/2004 | 205 | Nov 01/2003 |
| 19 | Mar 10/2004 | 10 | Nov 01/2003 | 206 | Nov 01/2003 |
| 20 | Mar 10/2004 | 11 | Mar 10/2004 | 207 | Jul 10/2005 |
| 21 | Mar 10/2004 | 12 | Mar 10/2004 | 208 | Jul 10/2004 |
| 22 | Mar 10/2004 | 13 | Nov 01/2003 | 57-53-71 IDENTIFICATION 1 | |
| 23 | Mar 10/2004 | 14 | Nov 10/2005 | 1 | Mar 10/2004 |
| 24 | Mar 10/2004 | 15 | Mar 10/2004 | 2 | Mar 10/2004 |
| 57-53-70 IDENTIFICATION 2 | | 16 | Nov 10/2006 | 3 | Mar 10/2004 |
| 1 | Mar 10/2009 | 17 | Nov 10/2006 | 4 | Nov 10/2006 |
| 2 | Mar 10/2004 | 18 | Nov 10/2006 | 5 | Mar 10/2004 |
| 3 | Jul 10/2004 | 19 | Nov 10/2006 | 6 | Mar 10/2004 |
| 4 | Mar 10/2004 | 20 | Nov 10/2006 | 7 | Mar 10/2004 |
| 5 | Mar 10/2004 | 21 | Nov 10/2006 | 8 | Nov 01/2003 |
| 6 | Mar 10/2004 | 22 | Nov 10/2006 | 57-53-71 ALLOWABLE DAMAGE 1 | |
| 7 | Mar 10/2004 | 23 | Nov 10/2006 | 101 | Nov 01/2003 |
| 8 | Mar 10/2004 | 24 | Nov 10/2006 | 102 | Nov 01/2003 |
| 9 | Nov 01/2003 | 25 | Nov 10/2006 | 103 | Nov 01/2003 |
| 10 | Mar 10/2004 | 26 | Nov 10/2006 | 104 | Jul 10/2004 |
| 11 | Mar 10/2004 | 27 | Nov 10/2006 | 105 | Nov 01/2003 |
| 12 | Nov 01/2003 | 28 | BLANK | 106 | Nov 01/2003 |
| 13 | Nov 01/2003 | 57-53-70 ALLOWABLE DAMAGE 1 | | 107 | Nov 01/2003 |
| 14 | Mar 10/2004 | 101 | Nov 01/2003 | 108 | BLANK |
| 15 | Mar 10/2004 | 102 | Nov 01/2003 | 57-53-71 REPAIR 1 | |
| 16 | Mar 10/2004 | 103 | Nov 01/2003 | 201 | Jul 10/2004 |
| 17 | Mar 10/2004 | 104 | Nov 01/2003 | 202 | Jul 10/2004 |
| 18 | Nov 01/2003 | 105 | Nov 10/2007 | 203 | Jul 10/2004 |
| 19 | Mar 10/2004 | 106 | Jul 10/2005 | 204 | Nov 10/2006 |
| 20 | Mar 10/2004 | 107 | Jul 10/2004 | 205 | Nov 10/2006 |
| 21 | Nov 01/2003 | 108 | Jul 10/2004 | 206 | Jul 10/2004 |
| 22 | BLANK | 109 | Jul 10/2004 | 57-60-00 GENERAL | |
| 57-53-70 IDENTIFICATION 3 | | 110 | Jul 10/2004 | 1 | Nov 01/2003 |
| 1 | Jul 10/2004 | 111 | Jul 10/2004 | 2 | BLANK |
| 2 | Mar 10/2004 | 112 | Jul 10/2004 | 57-60-01 IDENTIFICATION 1 | |
| 3 | Mar 10/2004 | 113 | Jul 10/2004 | 1 | Mar 10/2004 |
| 4 | Mar 10/2004 | 114 | BLANK | 2 | Mar 10/2004 |

A = Added, R = Revised, O = Overflow, D = Deleted

## CHAPTER 57
## WINGS

| Subject/Page | Date |
|---|---|
| 57-60-01 IDENTIFICATION 1 (cont) | |
| 3 | Nov 01/2003 |
| 4 | Jul 10/2004 |
| 5 | Mar 10/2004 |
| 6 | Mar 10/2004 |
| 7 | Mar 10/2004 |
| 8 | Nov 01/2003 |
| 9 | Jul 10/2004 |
| 10 | Mar 10/2004 |
| 11 | Nov 01/2003 |
| 12 | Jul 10/2004 |
| 13 | Mar 10/2004 |
| 14 | Mar 10/2004 |
| 15 | Mar 10/2004 |
| 16 | Mar 10/2004 |
| 17 | Mar 10/2004 |
| 18 | Mar 10/2004 |
| 19 | Mar 10/2004 |
| 20 | Nov 01/2003 |
| 57-60-01 IDENTIFICATION 2 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Mar 10/2004 |
| 5 | Mar 10/2004 |
| 6 | Mar 10/2004 |
| 7 | Mar 10/2004 |
| 8 | Mar 10/2004 |
| 9 | Mar 10/2004 |
| 10 | Mar 10/2004 |
| 11 | Mar 10/2004 |
| 12 | Nov 01/2003 |
| 57-60-01 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 10/2007 |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Nov 01/2003 |
| 106 | Nov 01/2003 |
| 107 | Nov 10/2007 |
| 57-60-01 ALLOWABLE DAMAGE 1 (cont) | |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 110 | Nov 10/2007 |
| 111 | Nov 10/2007 |
| 112 | Nov 01/2003 |
| 113 | Nov 01/2003 |
| 114 | BLANK |
| 57-60-01 ALLOWABLE DAMAGE 2 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Nov 10/2007 |
| 104 | Nov 01/2003 |
| 105 | Nov 10/2007 |
| 106 | Nov 10/2007 |
| 107 | Nov 01/2003 |
| 108 | Nov 10/2007 |
| 57-60-01 REPAIR 1 | |
| 201 | Nov 01/2003 |
| 202 | Jul 10/2005 |
| 203 | Nov 01/2003 |
| 204 | Nov 01/2003 |
| 205 | Nov 01/2003 |
| 206 | Nov 01/2003 |
| 207 | Nov 01/2003 |
| 208 | Jul 10/2005 |
| 209 | Jul 10/2005 |
| 210 | Nov 01/2003 |
| 211 | Jul 10/2004 |
| 212 | Jul 10/2004 |
| 213 | Jul 10/2004 |
| 214 | BLANK |
| 57-60-01 REPAIR 2 | |
| 201 | Nov 01/2003 |
| 202 | Jul 10/2005 |
| 203 | Nov 01/2003 |
| 204 | Jul 10/2005 |
| 205 | Jul 10/2004 |
| 206 | Jul 10/2004 |
| 57-60-02 IDENTIFICATION 1 | |
| 1 | Jul 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Nov 01/2003 |
| 4 | Mar 10/2004 |
| 5 | Mar 10/2004 |
| 6 | Mar 10/2004 |
| 7 | Mar 10/2004 |
| 8 | Mar 10/2004 |
| 9 | Mar 10/2004 |
| 10 | Mar 10/2004 |
| 11 | Mar 10/2004 |
| 12 | Mar 10/2004 |
| 13 | Mar 10/2004 |
| 14 | Mar 10/2004 |
| 15 | Mar 10/2004 |
| 16 | Mar 10/2004 |
| 17 | Mar 10/2004 |
| 18 | Mar 10/2004 |
| 19 | Mar 10/2004 |
| 20 | Nov 01/2003 |
| 57-60-02 IDENTIFICATION 2 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Nov 01/2003 |
| 4 | BLANK |
| 57-60-02 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Nov 10/2007 |
| 106 | Jul 10/2004 |
| 107 | Nov 01/2003 |
| 108 | Nov 10/2007 |
| 109 | Nov 10/2007 |
| 110 | Nov 10/2007 |
| 111 | Nov 10/2007 |
| 112 | Nov 10/2007 |
| 113 | Nov 10/2007 |

A = Added, R = Revised, O = Overflow, D = Deleted

# CHAPTER 57
# WINGS

| Subject/Page | Date |
|---|---|
| 57-60-02 ALLOWABLE DAMAGE 1 (cont) | |
| 114 | Nov 10/2007 |
| 57-60-02 ALLOWABLE DAMAGE 2 | |
| 101 | Nov 10/2007 |
| 102 | Nov 01/2003 |
| 103 | Jul 10/2004 |
| 104 | Jul 10/2004 |
| 105 | Jul 10/2004 |
| 106 | Jul 10/2004 |
| 107 | Jul 10/2004 |
| 108 | BLANK |
| 57-60-02 REPAIR 1 | |
| 201 | Nov 01/2003 |
| 202 | Nov 01/2003 |
| 203 | Nov 10/2007 |
| 204 | Nov 01/2003 |
| 205 | Nov 01/2003 |
| 206 | Nov 01/2003 |
| 207 | Nov 01/2003 |
| 208 | Nov 01/2003 |
| 209 | Nov 01/2003 |
| 210 | Nov 01/2003 |
| 211 | Nov 01/2003 |
| 212 | Nov 01/2003 |
| 213 | Nov 01/2003 |
| 214 | Nov 01/2003 |
| 215 | Nov 01/2003 |
| 216 | Nov 10/2007 |
| 217 | Jul 10/2004 |
| 218 | Jul 10/2004 |
| 219 | Jul 10/2004 |
| 220 | BLANK |
| 57-60-02 REPAIR 2 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-60-70 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 10/2006 |
| 102 | BLANK |
| 57-60-70 REPAIR 1 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-60-90 IDENTIFICATION 1 | |
| 1 | Jul 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Nov 01/2003 |
| 5 | Nov 01/2003 |
| 6 | BLANK |
| 57-60-90 IDENTIFICATION 2 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Nov 01/2003 |
| 4 | BLANK |
| 57-60-90 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Nov 01/2003 |
| 106 | Jul 10/2004 |
| 107 | Nov 01/2003 |
| 108 | Nov 01/2003 |
| 109 | Nov 01/2003 |
| 110 | BLANK |
| 57-60-90 ALLOWABLE DAMAGE 2 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Jul 10/2004 |
| 104 | Nov 01/2003 |
| 105 | Nov 01/2003 |
| 106 | BLANK |
| 57-60-90 REPAIR 1 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-60-90 REPAIR 2 | |
| 201 | Nov 10/2006 |
| 202 | BLANK |
| 57-70-00 GENERAL | |
| 1 | Nov 01/2003 |
| 2 | BLANK |
| 57-70-01 IDENTIFICATION 1 | |
| 1 | Jul 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Mar 10/2004 |
| 57-70-01 IDENTIFICATION 2 | |
| 1 | Mar 10/2004 |
| 2 | Mar 10/2004 |
| 3 | Mar 10/2004 |
| 4 | Mar 10/2004 |
| 5 | Mar 10/2004 |
| 6 | Mar 10/2004 |
| 7 | Mar 10/2004 |
| 8 | Mar 10/2004 |
| 9 | Mar 10/2004 |
| 10 | Mar 10/2004 |
| 11 | Mar 10/2004 |
| 12 | Mar 10/2004 |
| 13 | Mar 10/2004 |
| 14 | Mar 10/2004 |
| 15 | Mar 10/2004 |
| 16 | Mar 10/2004 |
| 17 | Mar 10/2004 |
| 18 | BLANK |
| 57-70-01 ALLOWABLE DAMAGE 1 | |
| 101 | Nov 01/2003 |
| 102 | Nov 01/2003 |
| 103 | Nov 01/2003 |
| 104 | Nov 01/2003 |
| 105 | Nov 01/2003 |
| 106 | Nov 01/2003 |
| 107 | Nov 01/2003 |
| 108 | Nov 10/2006 |
| 109 | Jul 10/2008 |
| 110 | Jul 10/2008 |
| 111 | Nov 10/2006 |
| 112 | Nov 10/2006 |

A = Added, R = Revised, O = Overflow, D = Deleted

# CHAPTER 57
# WINGS

| Subject/Page | Date | Subject/Page | Date | Subject/Page | Date |
|---|---|---|---|---|---|
| 57-70-01 ALLOWABLE DAMAGE 1 (cont) | | 57-70-02 ALLOWABLE DAMAGE 1 (cont) | | | |
| 113 | Nov 01/2003 | 107 | Nov 01/2003 | | |
| 114 | Nov 01/2003 | 108 | Nov 01/2003 | | |
| 115 | Nov 01/2003 | 109 | Mar 10/2005 | | |
| 116 | Nov 01/2003 | 110 | Nov 01/2003 | | |
| 117 | Nov 01/2003 | 111 | Nov 01/2003 | | |
| 118 | Nov 01/2003 | 112 | Nov 01/2003 | | |
| 119 | Nov 01/2003 | 57-70-02 REPAIR 1 | | | |
| 120 | BLANK | 201 | Nov 10/2006 | | |
| 57-70-01 REPAIR 1 | | 202 | Nov 01/2003 | | |
| 201 | Nov 10/2006 | 203 | Nov 10/2007 | | |
| 202 | Nov 01/2003 | 204 | Nov 10/2007 | | |
| 203 | Mar 10/2005 | 205 | Nov 01/2003 | | |
| 204 | BLANK | 206 | BLANK | | |
| 57-70-02 IDENTIFICATION 1 | | 57-70-90 ALLOWABLE DAMAGE 1 | | | |
| 1 | Jul 10/2004 | 101 | Nov 01/2003 | | |
| 2 | Mar 10/2004 | 102 | BLANK | | |
| 3 | Nov 01/2003 | 57-70-90 REPAIR 1 | | | |
| 4 | BLANK | 201 | Nov 10/2006 | | |
| 57-70-02 IDENTIFICATION 2 | | 202 | BLANK | | |
| 1 | Mar 10/2004 | | | | |
| 2 | Mar 10/2004 | | | | |
| 3 | Mar 10/2004 | | | | |
| 4 | Mar 10/2004 | | | | |
| 5 | Mar 10/2004 | | | | |
| 6 | Mar 10/2004 | | | | |
| 7 | Mar 10/2004 | | | | |
| 8 | Mar 10/2004 | | | | |
| 9 | Mar 10/2004 | | | | |
| 10 | Mar 10/2004 | | | | |
| 11 | Mar 10/2004 | | | | |
| 12 | Nov 01/2003 | | | | |
| 57-70-02 ALLOWABLE DAMAGE 1 | | | | | |
| 101 | Nov 01/2003 | | | | |
| 102 | Mar 10/2005 | | | | |
| 103 | Nov 01/2003 | | | | |
| 104 | Nov 01/2003 | | | | |
| 105 | Nov 01/2003 | | | | |
| 106 | Nov 01/2003 | | | | |

A = Added, R = Revised, O = Overflow, D = Deleted

# CHAPTER 57
# WINGS

# CHAPTER 57
# WINGS

## CHAPTER 57
## WINGS

# CHAPTER 57
# WINGS

CHAPTER 57
WINGS

## CHAPTER 57
## WINGS

CHAPTER 57
WINGS

# CHAPTER 57
# WINGS

# CHAPTER 57
# WINGS

## GENERAL - WINGS

### 1. General

A. This chapter contains information on identification, allowable damage, and repairs to the structural components of the wing, leading edge slats, trailing edge flaps, and the ailerons.

### 2. References

| Reference | Title |
|---|---|
| 51-20-05, GENERAL | Repair Sealing |
| 51-60-00, GENERAL | Control Surface Balance Procedures |

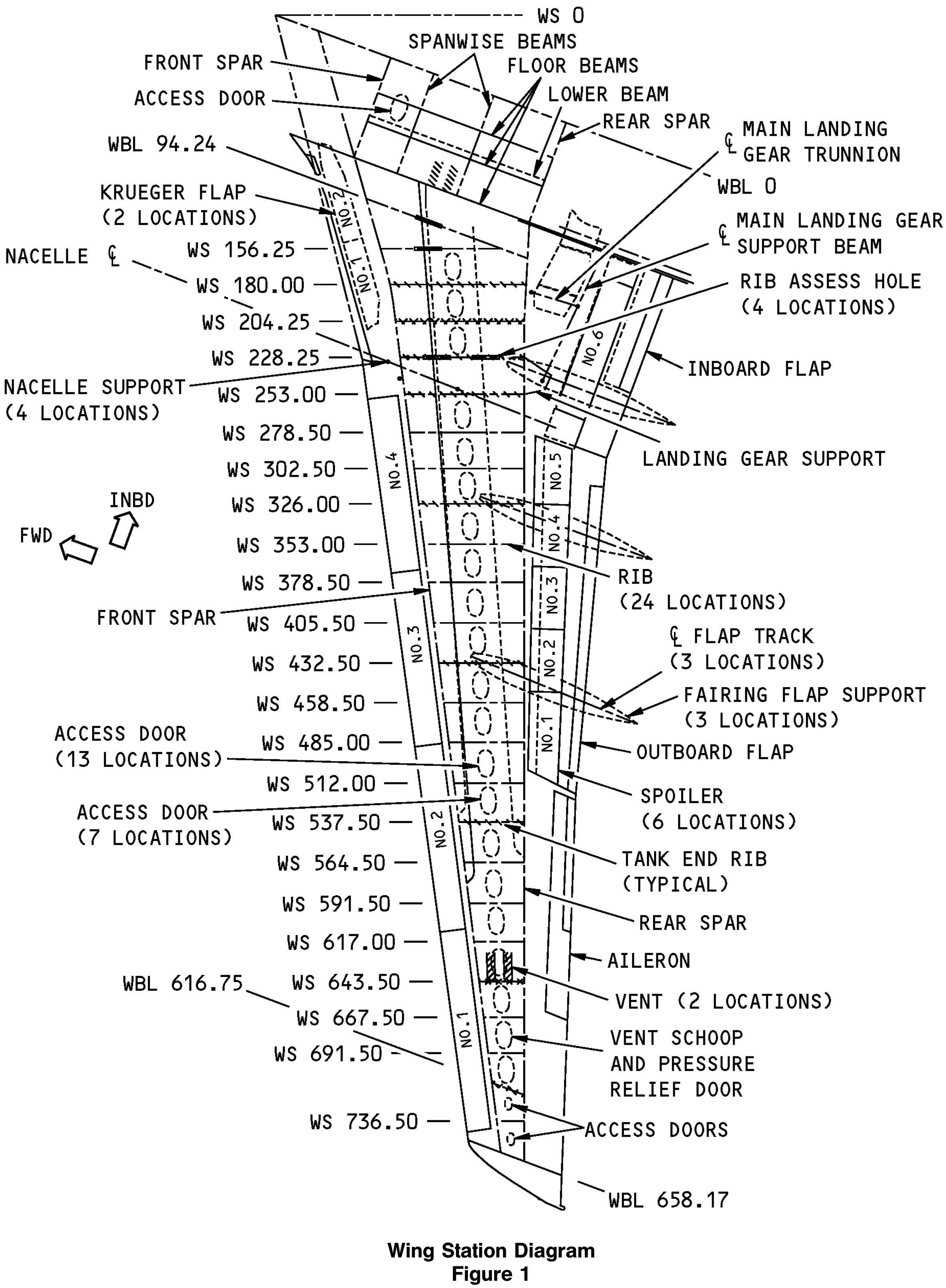
### 3. Sealing

A. The outer wing and center wing structure between the front and rear spars is sealed to from integral fuel tanks.

B. Repairs to the wing involving the fuel tank area must be sealed. Refer to 51-20-05, GENERAL for sealing requirements and processes.

### 4. Control Surface Balancing

A. Refer to 51-60-00, GENERAL for general information on control surface balancing.

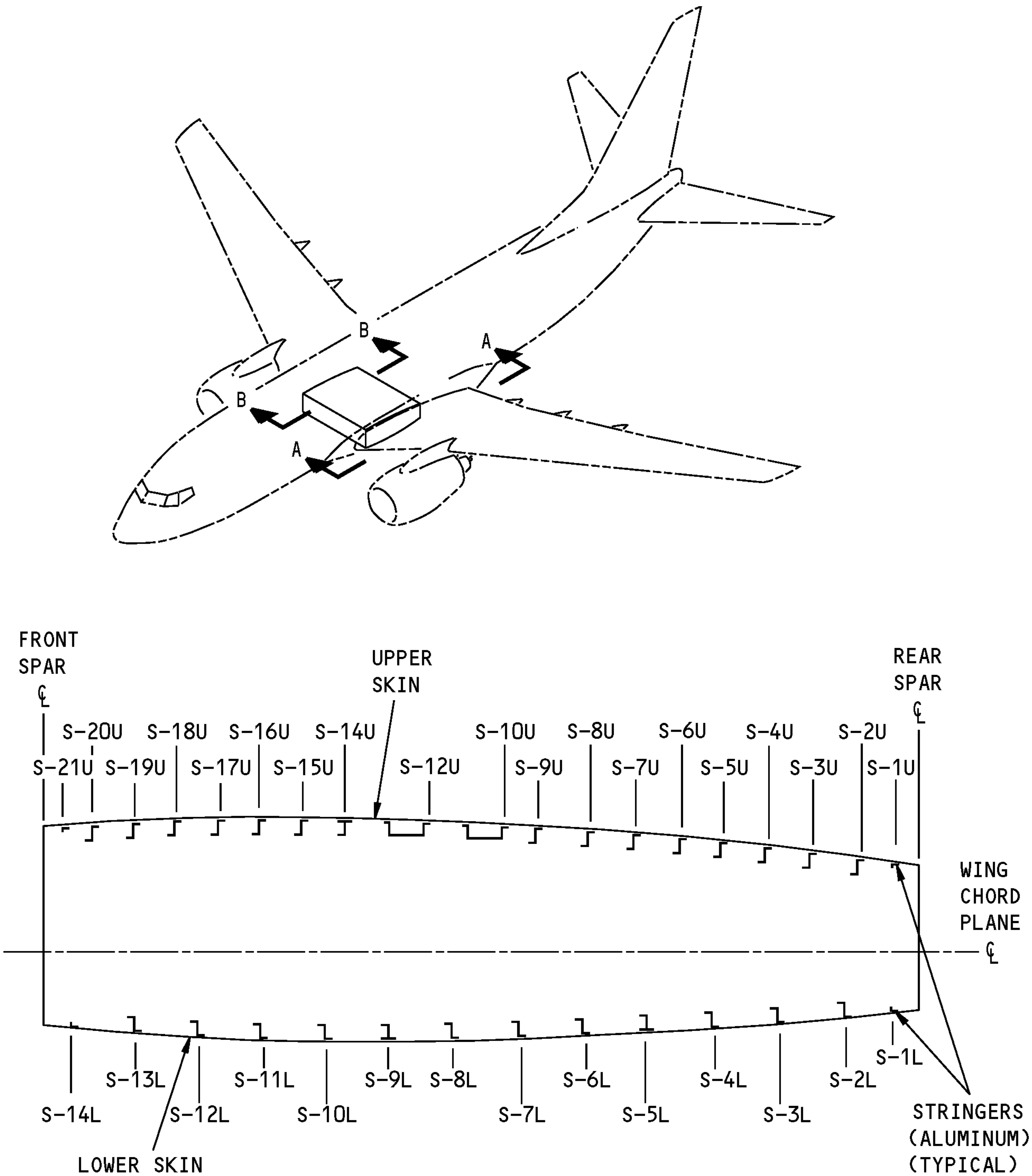



**Wing Station Diagram**
**Figure 1**

## ALLOWABLE DAMAGE 1 - WING STRINGERS

### 1. Applicability

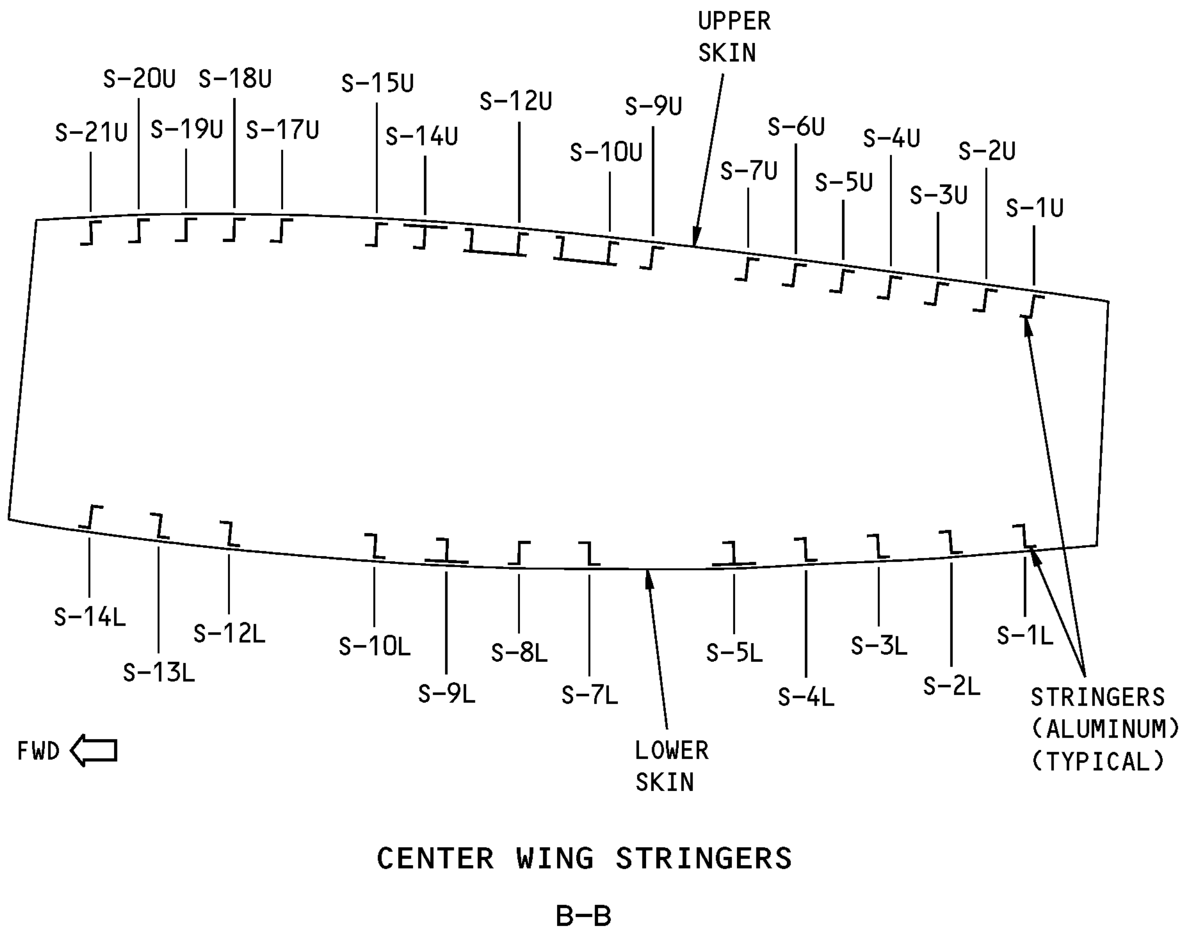
A. This subject gives the allowable damage limits for the center and outer wing stringers shown in Wing Stringer Locations, Figure 101/ALLOWABLE DAMAGE 1.




OUTER WING STRINGERS

A-A

**Wing Stringer Locations**
**Figure 101 (Sheet 1 of 2)**




Wing Stringer Locations
Figure 101 (Sheet 2 of 2)

**2. General**

A. Refer to Wing Stringers, Figure 102/ALLOWABLE DAMAGE 1 for the wing stringer shapes.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

(1) Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limits that is applicable to each type of stringer.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| **TYPE OF STRINGER** | **PARAGRAPH** |
| Zee Stringers | 4.A |
| Splice Stringers | 4.B |
| Vent Stringers | 4.C |
| Outer Wing Stringer S-1U | 4.D |
| Outer Wing Stinger S-21U | 4.D |
| Outer Wing Stinger S-1L | 4.D |
| Outer Wing Stringer S-14L and S-1L | 4.D |

C. After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE ROTARY PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Rotary peen or shot peen the reworked areas. Refer to SOPM 20-10-03.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(3) Do the step that follows for the interior surfaces.

(a) Apply one layer of BMS 10-20, Type II primer to the conversion coated, reworked area of the stringer. Refer to AMM 28-11-00/701.

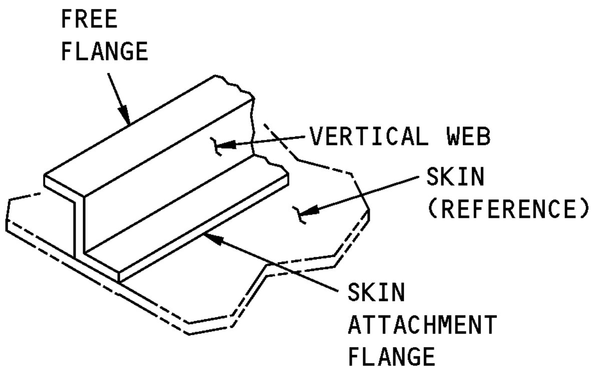
(4) Do the steps that follow for the exterior surfaces.

(a) Fill the space caused from the removal of damaged material at a skin splice with BMS 5-95 sealant. Refer to 51-20-05.

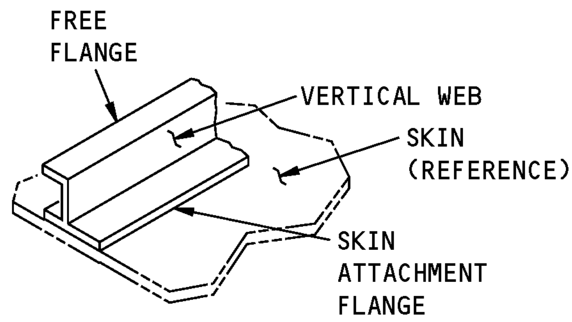
(b) Apply BMS 10-20, Type II primer to the reworked area. Refer to AMM 28-11-00/701.

(c) Apply BMS 10-20, Type II primer to the exterior surface of the BMS 5-95 sealant that was used to fill the space on the outer surface. Refer to SOPM 20-44-04.

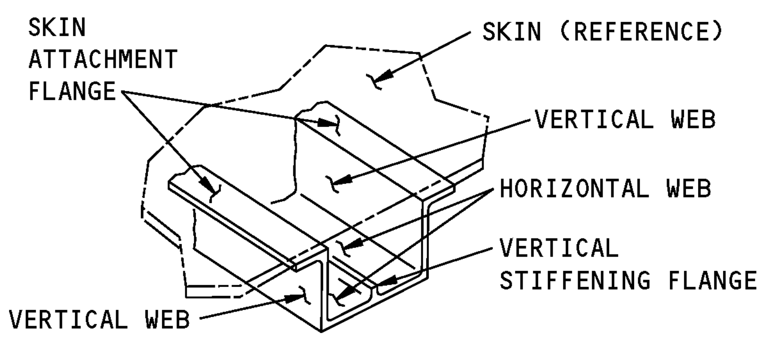
(d) Apply an applicable finish to the BMS 10-20, Type II primer. Refer to AMM 51-21-00/701.




ZEE STRINGER




SKIN SPLICE STRINGER

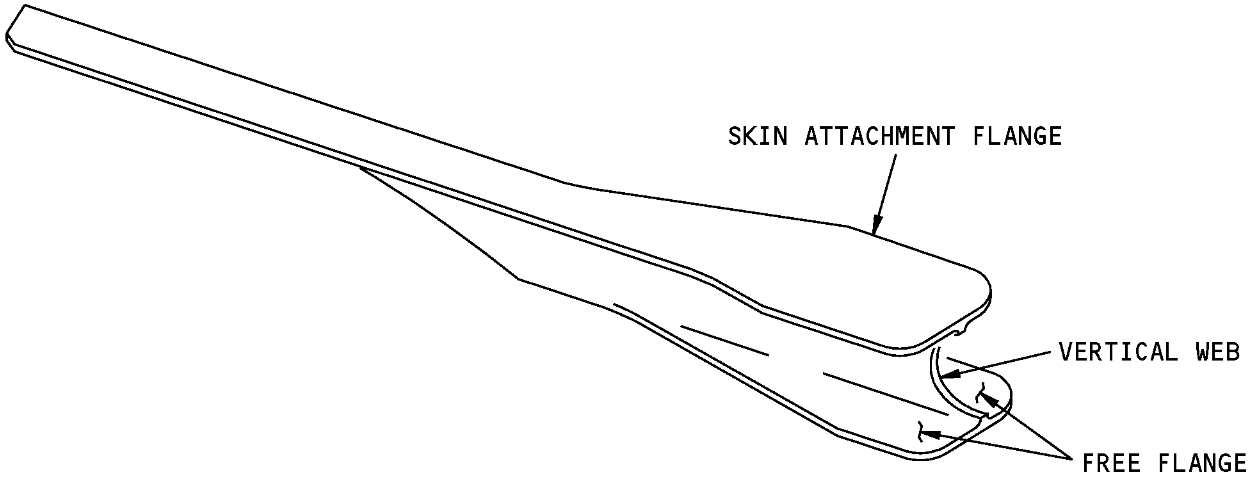



VENT STRINGER

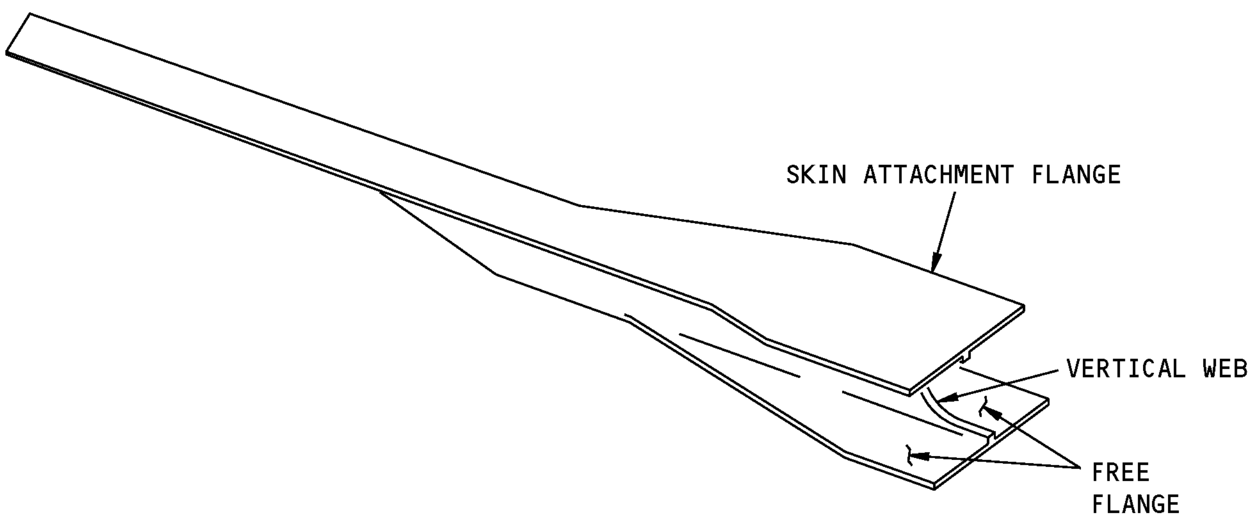
NOTES

- TYPICAL ALUMINUM STRINGER SECTIONS ARE SHOWN.

**Wing Stringers**
**Figure 102 (Sheet 1 of 3)**

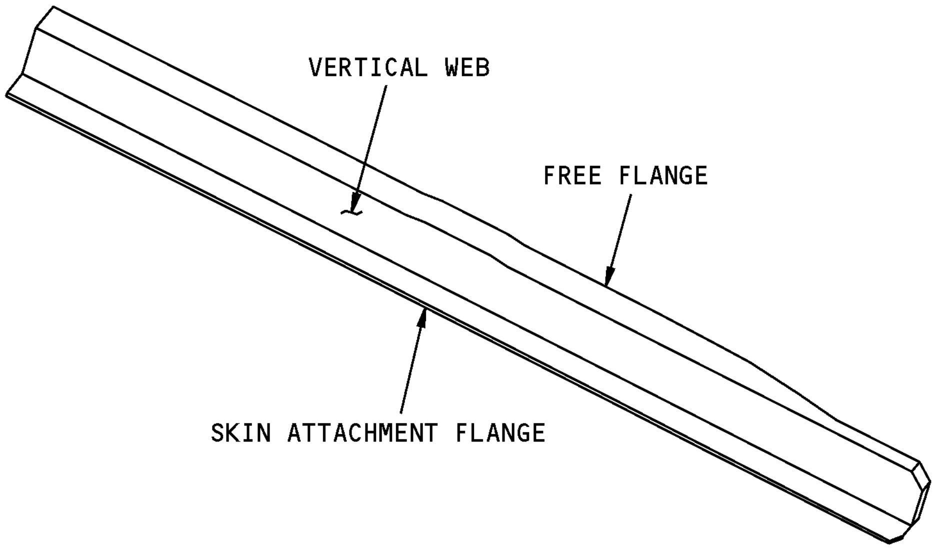



OUTER WING STRINGER S-1U

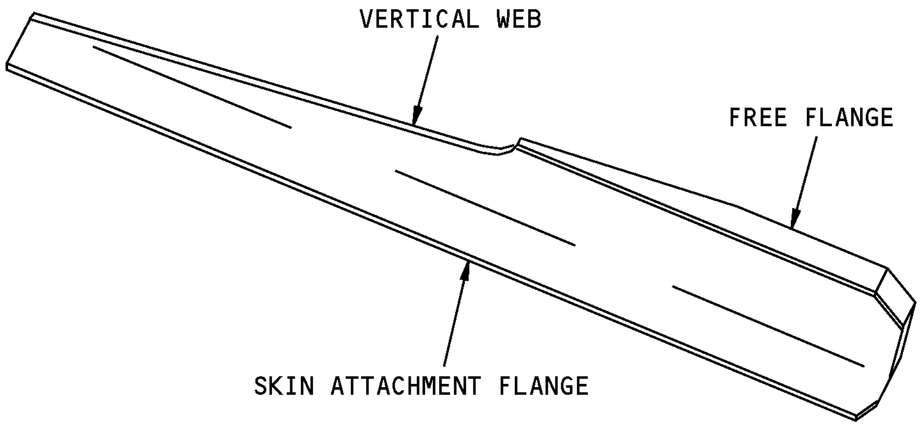



OUTER WING STRINGER S-21U

**Wing Stringers**
**Figure 102 (Sheet 2 of 3)**




OUTER WING STRINGER S-1L




OUTER WING STRINGER S-14L

**Wing Stringers**
**Figure 102 (Sheet 3 of 3)**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-09 | COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Zee Stringers

(1) Free Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

(2) Skin Attachment Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and G .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(3) Vertical Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

B. Skin Splice Stringers

(1) Free Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , C , D , E , and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

(2) Skin Attachment Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , G and H .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(3) Vertical Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

C. Vent Stringers

(1) Skin Attachment Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , C , D , E , and G .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Vertical Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a MS21141-()()P blind bolt installed wet with BMS 5-26 or BMS 5-45 sealant.

(3) Horizontal Web

(a) Cracks:

1) Remove the damage at the fuel system cutouts as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details I and J .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details E , G , I , and J .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(4) Vertical Stiffening Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail K .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details K and L .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

D. Outer Wing Stringers S-1U, S-21U, S-1L, and S-14L

(1) Free Flanges

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details C , D , E , and F .

(c) Dents are not permitted.

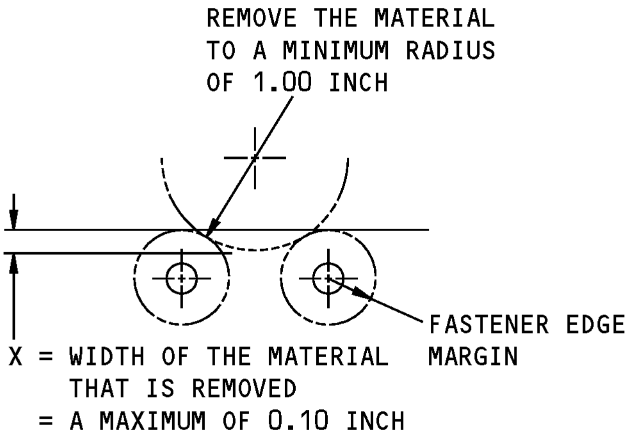
(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter
3) They are a minimum of 1.0 inch away from the edge of a hole or other damage
4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge
5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09
6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

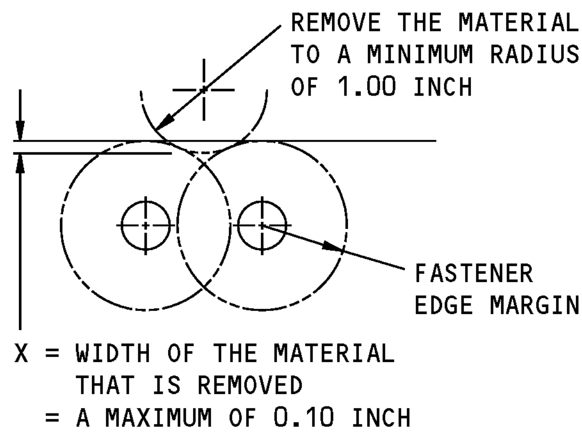
(2) Skin Attachment Flanges
(a) Cracks:
1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .
(b) Nicks, Gouges, Scratches, and Corrosion:
1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and G .
(c) Dents are not permitted.
(d) Holes and Punctures are not permitted.

(3) Vertical Web
(a) Cracks are not permitted.
(b) Nicks, Gouges, Scratches, and Corrosion:
1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details E and F .
(c) Dents are not permitted.
(d) Holes and Punctures are permitted if:
1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F
2) They are a maximum of 0.25 inch in diameter
3) They are a minimum of 1.0 inch away from the edge of a hole or other damage
4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge
5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09
6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.



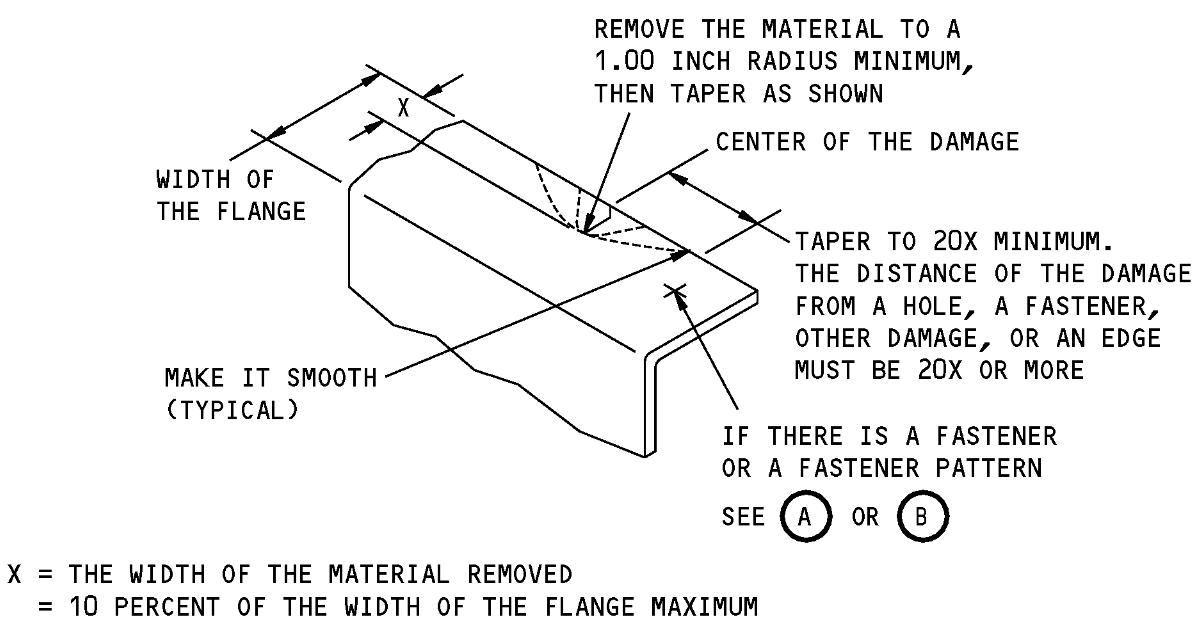

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

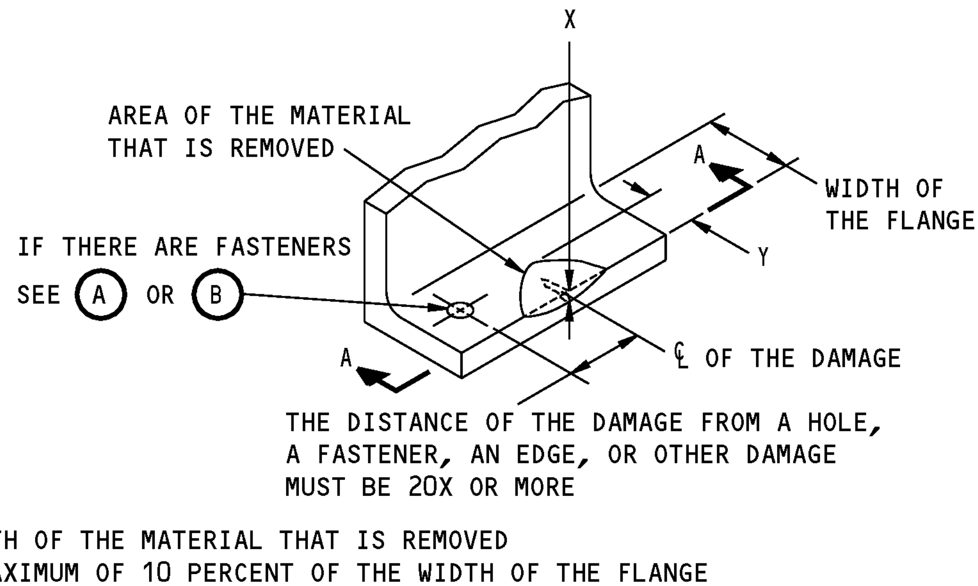
B



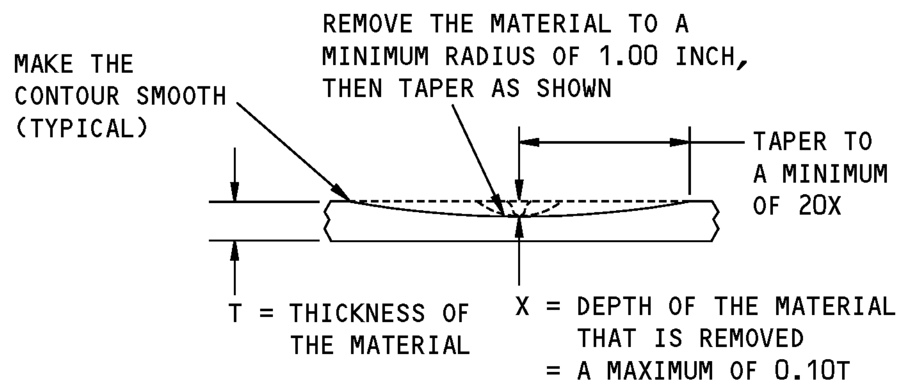

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 8)**

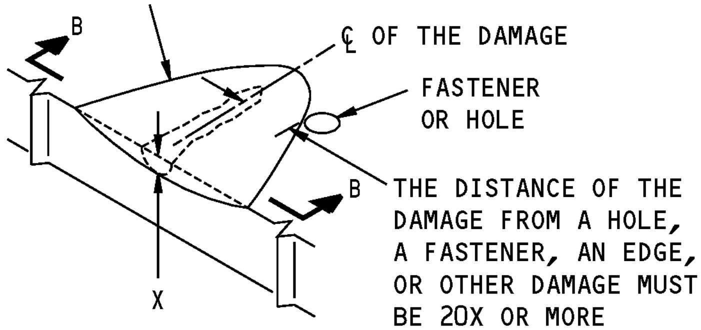



REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE
D



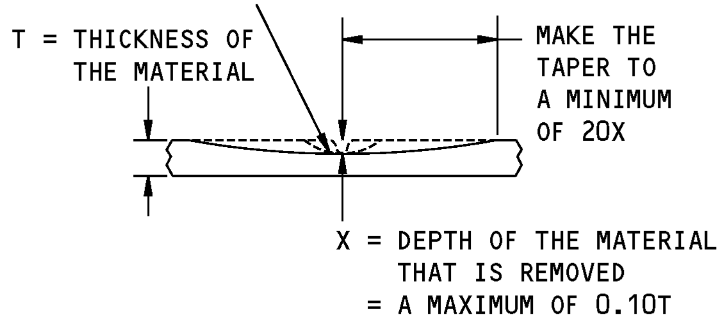

A-A

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 8)**



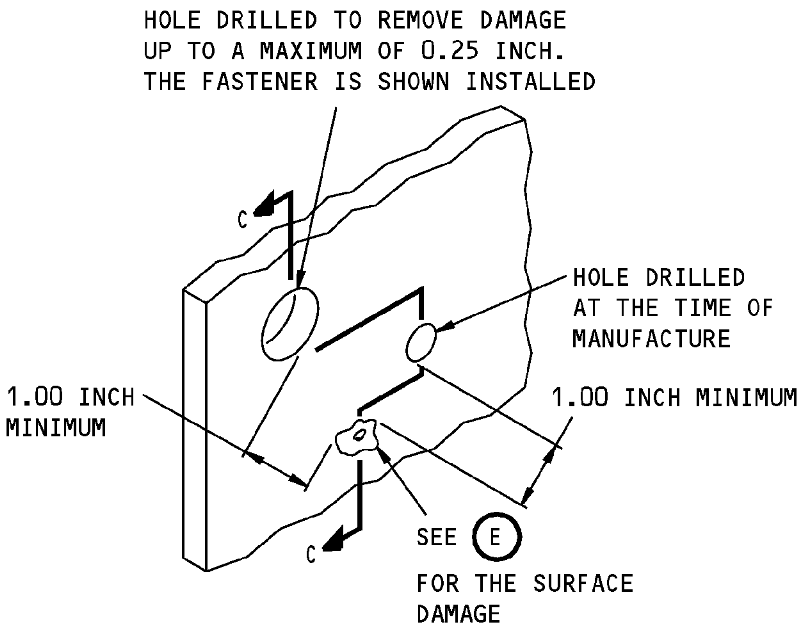

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

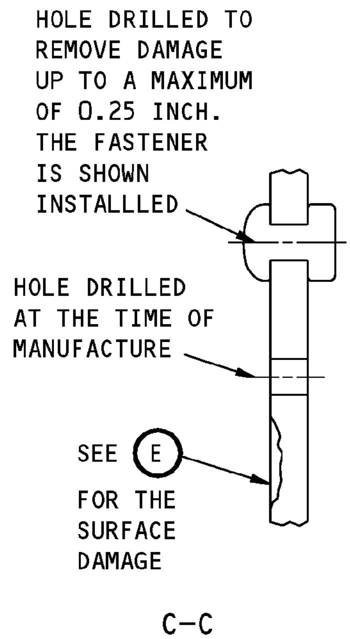
E




B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 8)**




$A_1$ = NET AREA = TOTAL FLANGE OR WEB AREA - HOLES MADE AT THE TIME OF MANUFACTURE.

$A_2$ = AREA REMOVED FOR NICKS, SCRATCHES, GOUGES, CORROSION, HOLES, AND PUNCTURES.

THE MAXIMUM LOSS PERMITTED IN THE CROSS-SECTIONAL AREA IS DEFINED AS FOLLOWS:

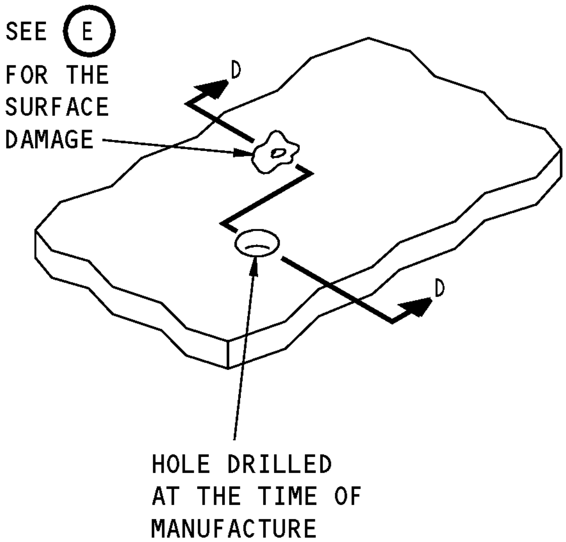
$$\frac{A_2}{A_1} \le 0.10$$

THE MAXIMUM LOSS IN THE CROSS-SECTIONAL AREA IS FOR DAMAGE THAT IS IN ALL 1.5 INCH LENGTHS ALONG THE STRINGER.

REMOVAL OF DAMAGED MATERIAL
IN A STRINGER FLANGE OR WEB

F

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 8)**




$A_1$ = NET FLANGE AREA = TOTAL FLANGE AREA − HOLES MADE AT THE TIME OF MANUFACTURE.

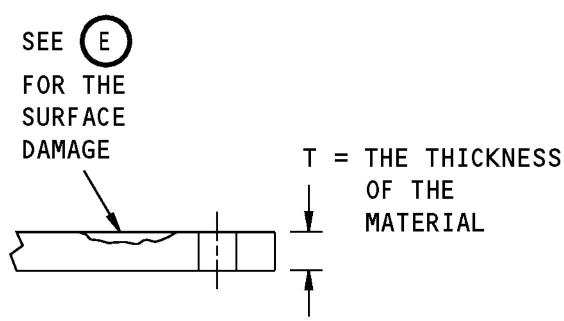
$A_2$ = AREA REMOVED FOR NICKS, SCRATCHES, GOUGES, AND CORROSION.

THE MAXIMUM LOSS PERMITTED IN THE CROSS-SECTIONAL AREA IS DEFINED AS FOLLOWS:

$$\frac{A_2}{A_1} \leq 0.10$$

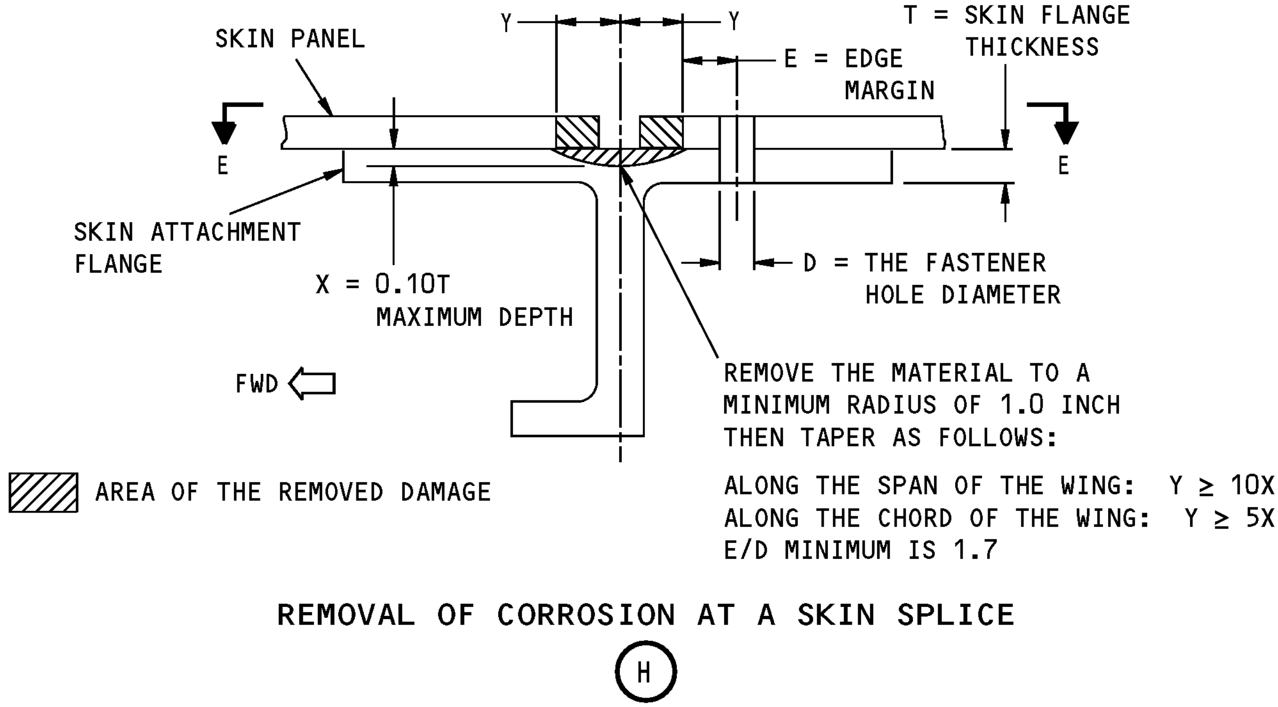
THE MAXIMUM LOSS IN THE CROSS-SECTIONAL AREA IS FOR DAMAGE THAT IS IN ALL 1.5 INCH LENGTHS ALONG THE STRINGER.

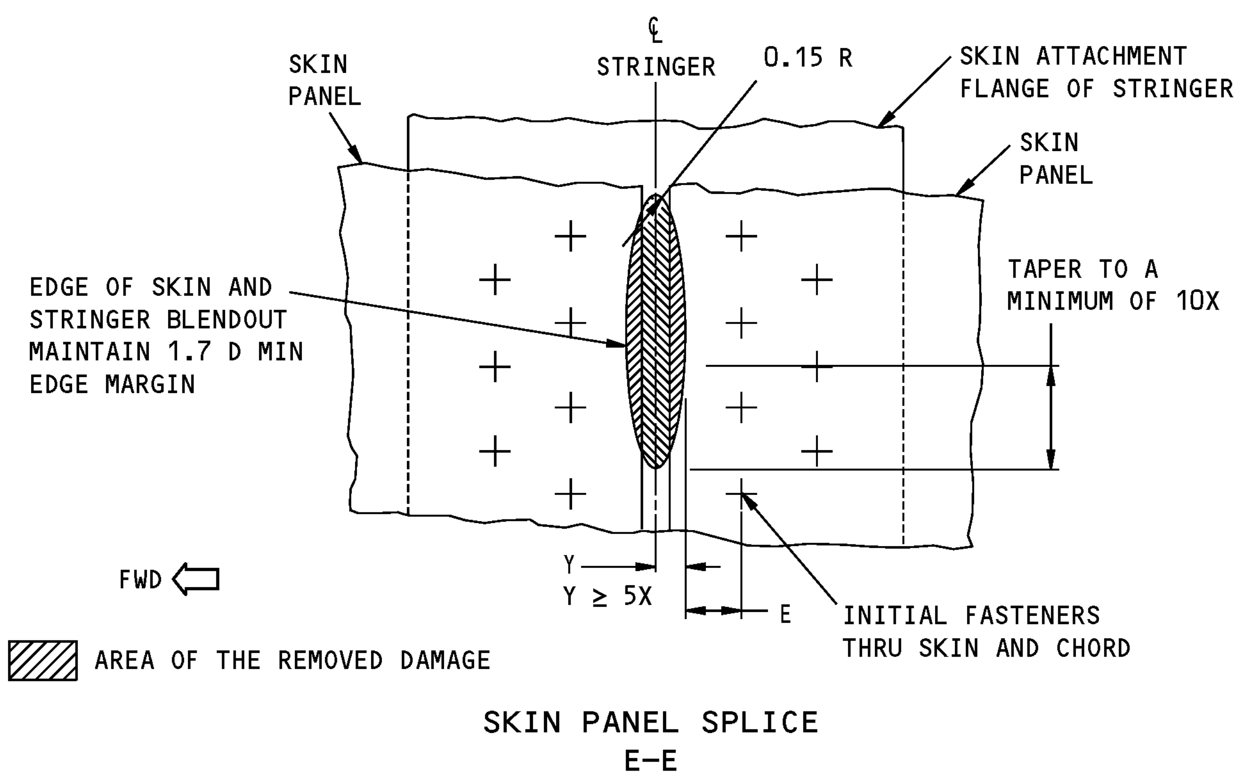
REMOVAL OF DAMAGED MATERIAL IN A STRINGER FLANGE
G



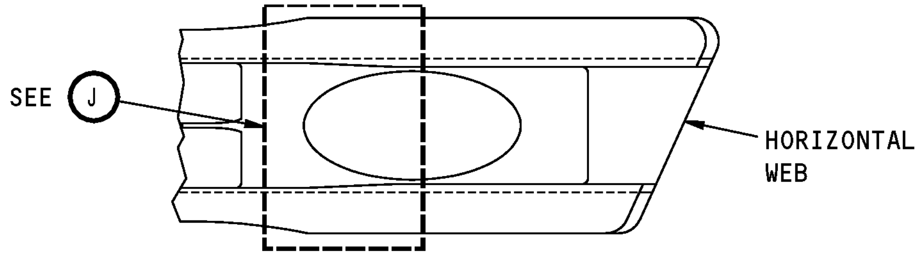

D-D

**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 8)**



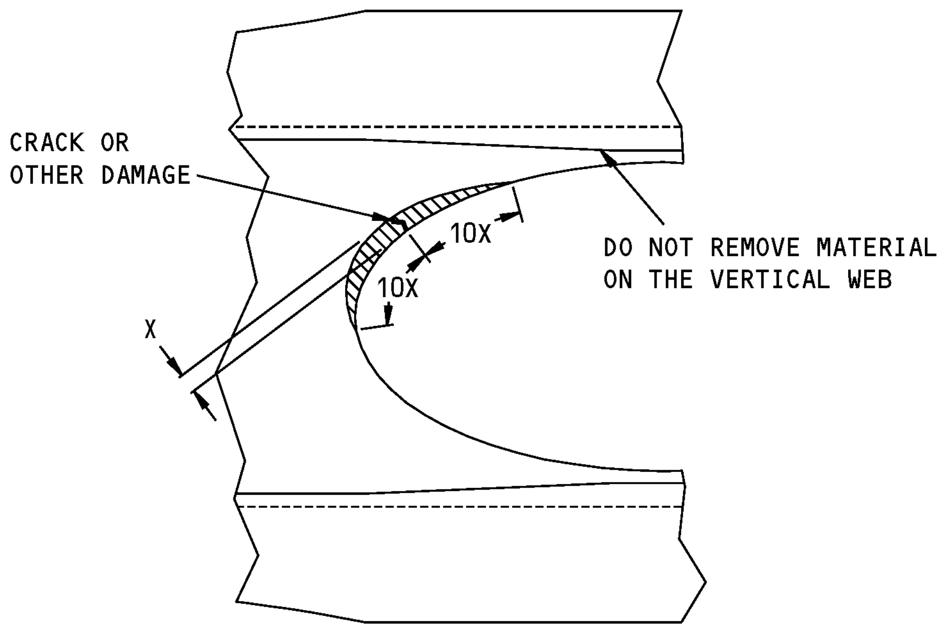




**Allowable Damage Limits**
**Figure 103 (Sheet 6 of 8)**




FUEL SYSTEM CUTOUT IN VENT STRINGER

I



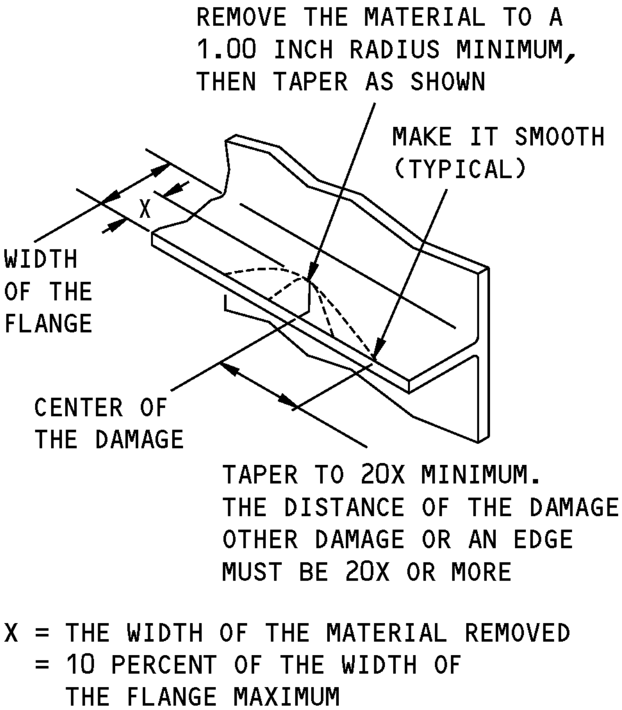

AREA OF THE REMOVED DAMAGE

X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

DAMAGE REMOVAL AT THE EDGE OF A FUEL SYSTEM CUTOUT
OVAL SHAPE HOLE IS SHOWN, DAMAGE REMOVAL ON THE CIRCULAR FUEL SYSTEM CUTOUT IS PERMITTED

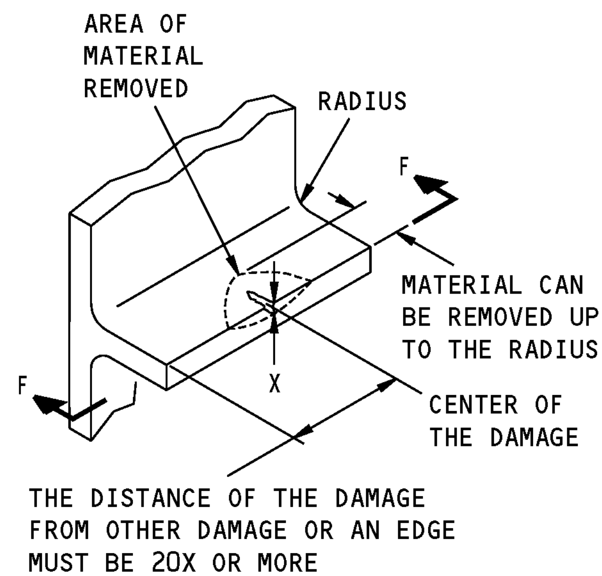
J

**Allowable Damage Limits**
**Figure 103 (Sheet 7 of 8)**



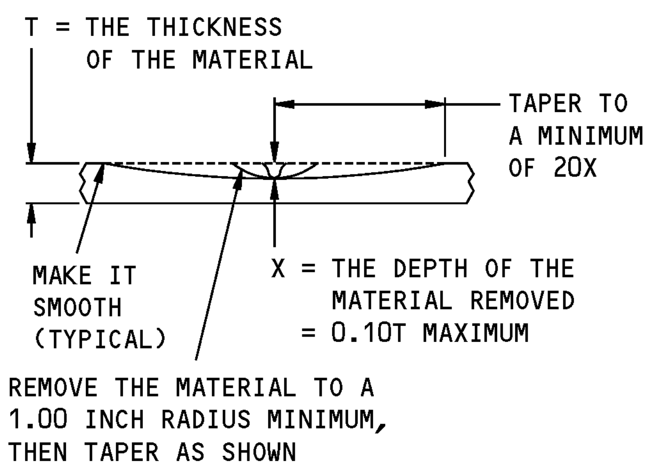

REMOVAL OF DAMAGED
MATERIAL ON AN EDGE

K




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

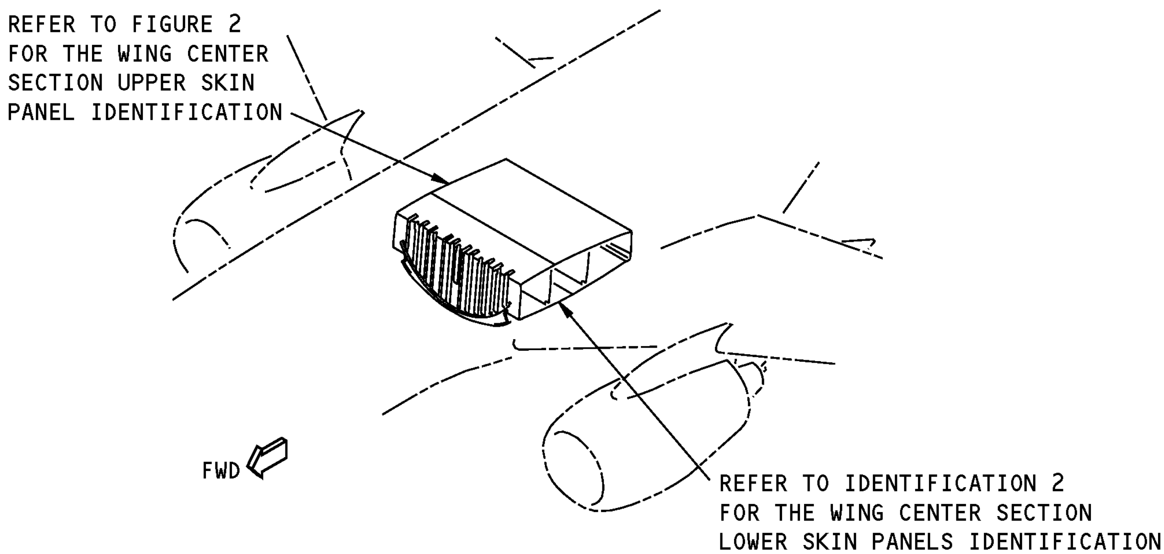
L




F-F

**Allowable Damage Limits**
**Figure 103 (Sheet 8 of 8)**

## IDENTIFICATION 1 - WING CENTER SECTION UPPER SKIN IDENTIFICATION

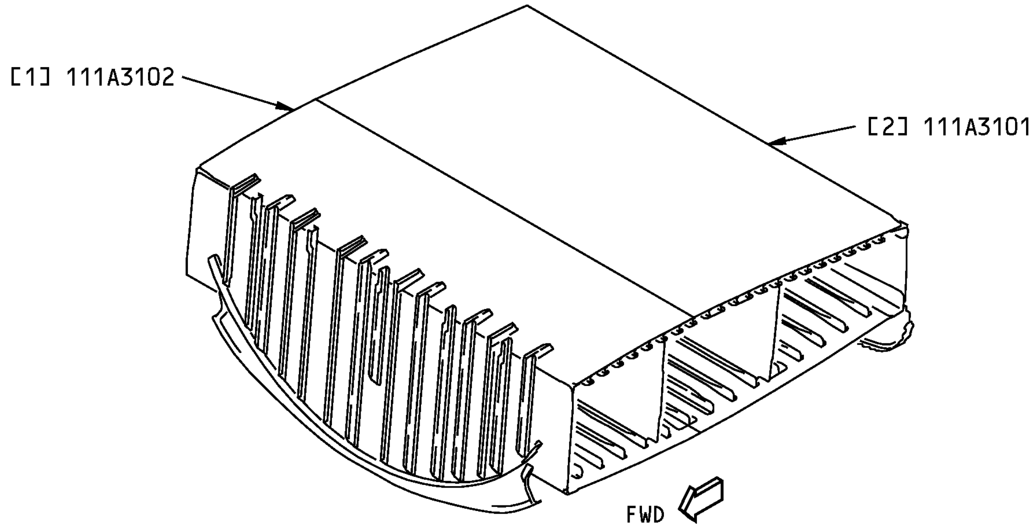



NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Center Section Upper Skin Panel Locations**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 111A3000 | Wing Center Section Upper Skin Panel Installation |
| 111A3101 | Wing Center Section Upper Aft Skin Panel |
| 111A3102 | Wing Center Section Upper Forward Skin Panel |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

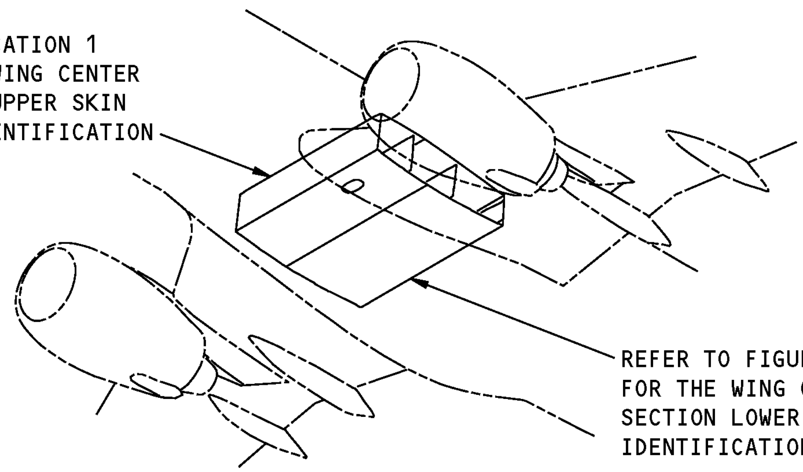
**Wing Center Section Upper Skin Panels Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Upper Forward Skin Panel | 0.375 (9.53) | 7055-T7751 plate as given in BMS 7-307 | |
| [2] | Upper Aft Skin Panel | 0.375 (9.53) | 7055-T7751 plate as given in BMS 7-307 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - WING CENTER SECTION LOWER SKIN IDENTIFICATION

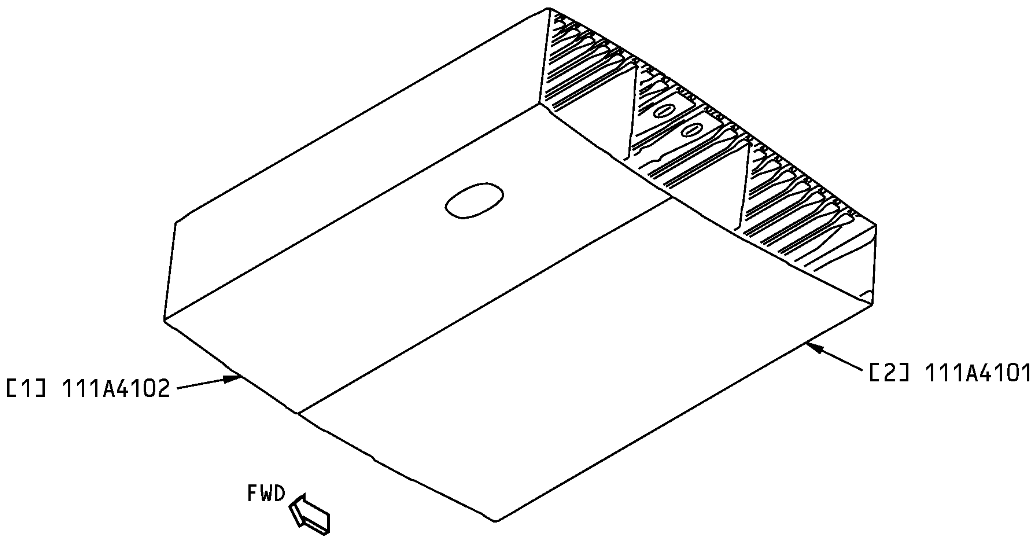



NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Center Section Lower Skin Panel Locations**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 111A4000 | Wing Center Section Lower Skin Panel Installation |
| 111A4101 | Wing Center Section Aft Lower Skin Panel |
| 111A4102 | Wing Center Section Forward Lower Skin Panel |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Wing Center Section Lower Skin Panels Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T**[*[1]] | **MATERIAL** | **EFFECTIVITY** |
| [1] | Forward Lower Skin Panel | 0.75 (19.05) | 2324-T39 plate as given in BMS 7-254 | |
| [2] | Aft Lower Skin Panel | 0.75 (19.05) | 2324-T39 plate as given in BMS 7-254 | |

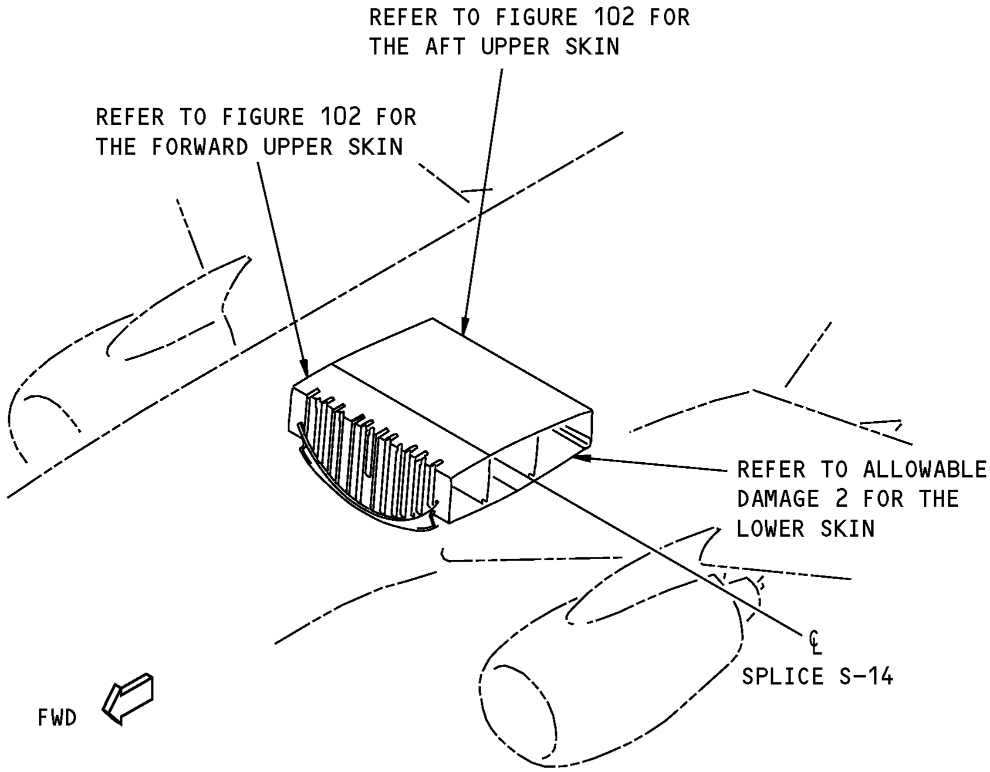
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## ALLOWABLE DAMAGE 1 - WING CENTER SECTION UPPER SKIN PANELS

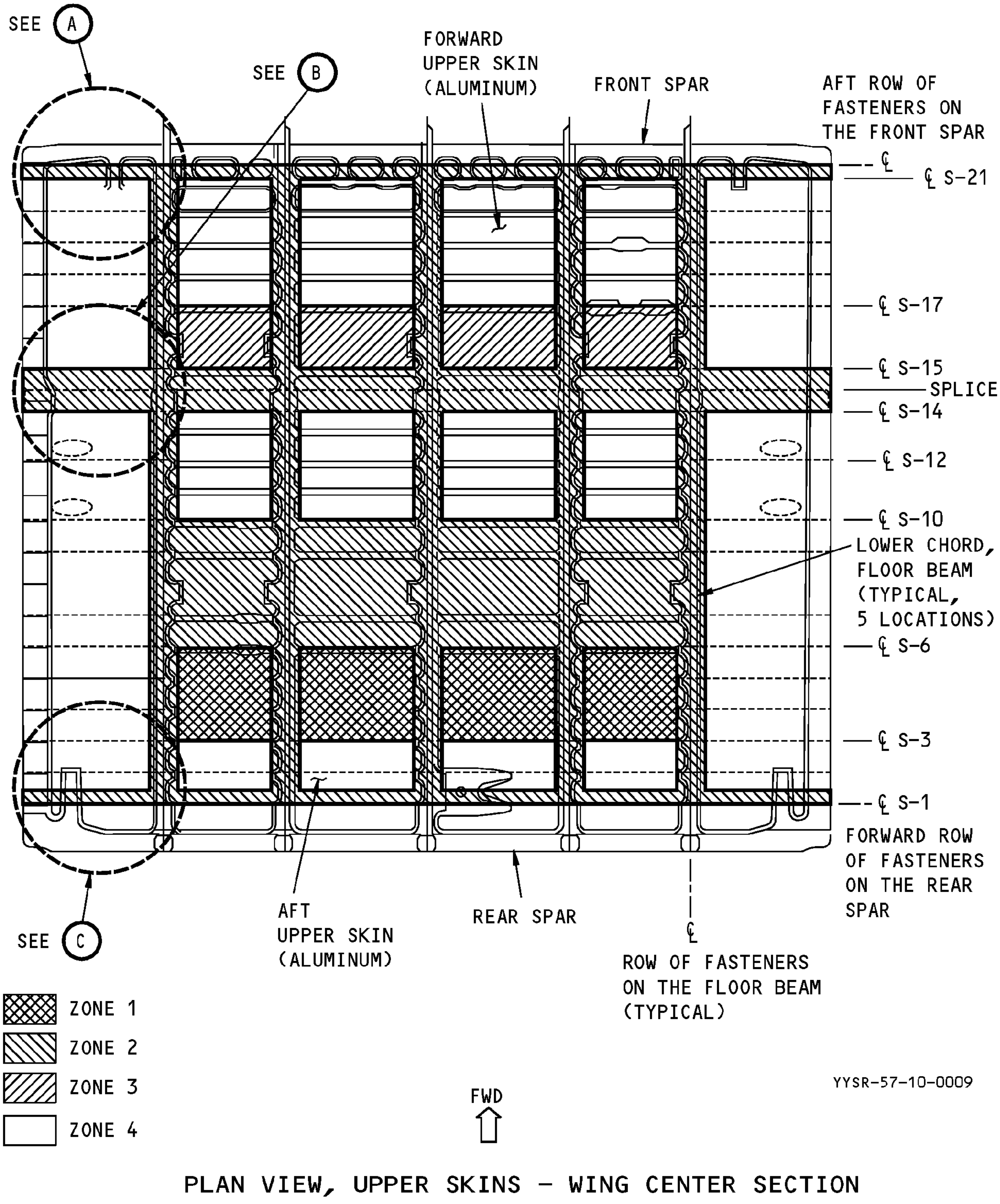
1. **Applicability**
   A. This subject gives the allowable damage limits for the upper skin panels of the wing center section shown in Wing Center Section Upper Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.
   B. Allowable Damage 1 is applicable to airplanes that have not had winglets installed.
   C. Allowable Damage 1 is not applicable to:
      (1) Airplanes that have had winglets installed as part of the production of the airplane. Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets.
      (2) Airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.
2. **General**
   A. Remove the damage as necessary.
      (1) Refer to 51-10-02 for the inspection and removal of damage.
      (2) Refer to 51-30-03 for the possible sources of the abrasive materials you can use to remove the damage.
      (3) Refer to 51-30-05 for the possible sources of the equipment and tools you can use to remove the damage.
   B. After you remove the damage, do the steps that follow:

      **WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

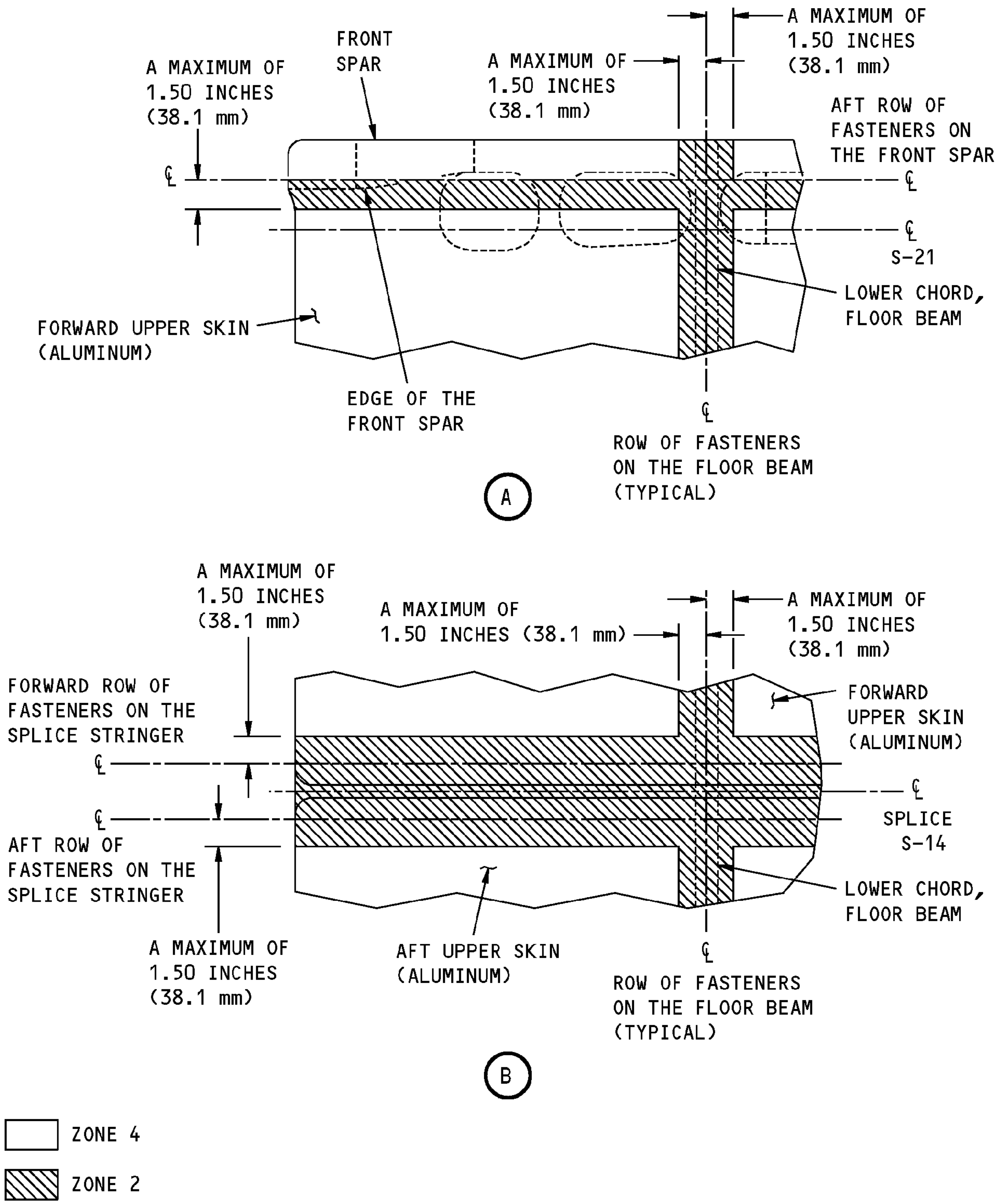
      (1) Flap peen or shot peen the reworked areas of the upper skin panels.
         (a) Refer to 51-20-06 for the shot peen intensity and shot number.
         (b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.
      (2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.
      (3) Apply a layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM 51-21-00/701.
      (4) Apply BMS 5-81, Type II secondary fuel barrier sealant to the exterior surfaces. Refer to AMM 28-11-00.



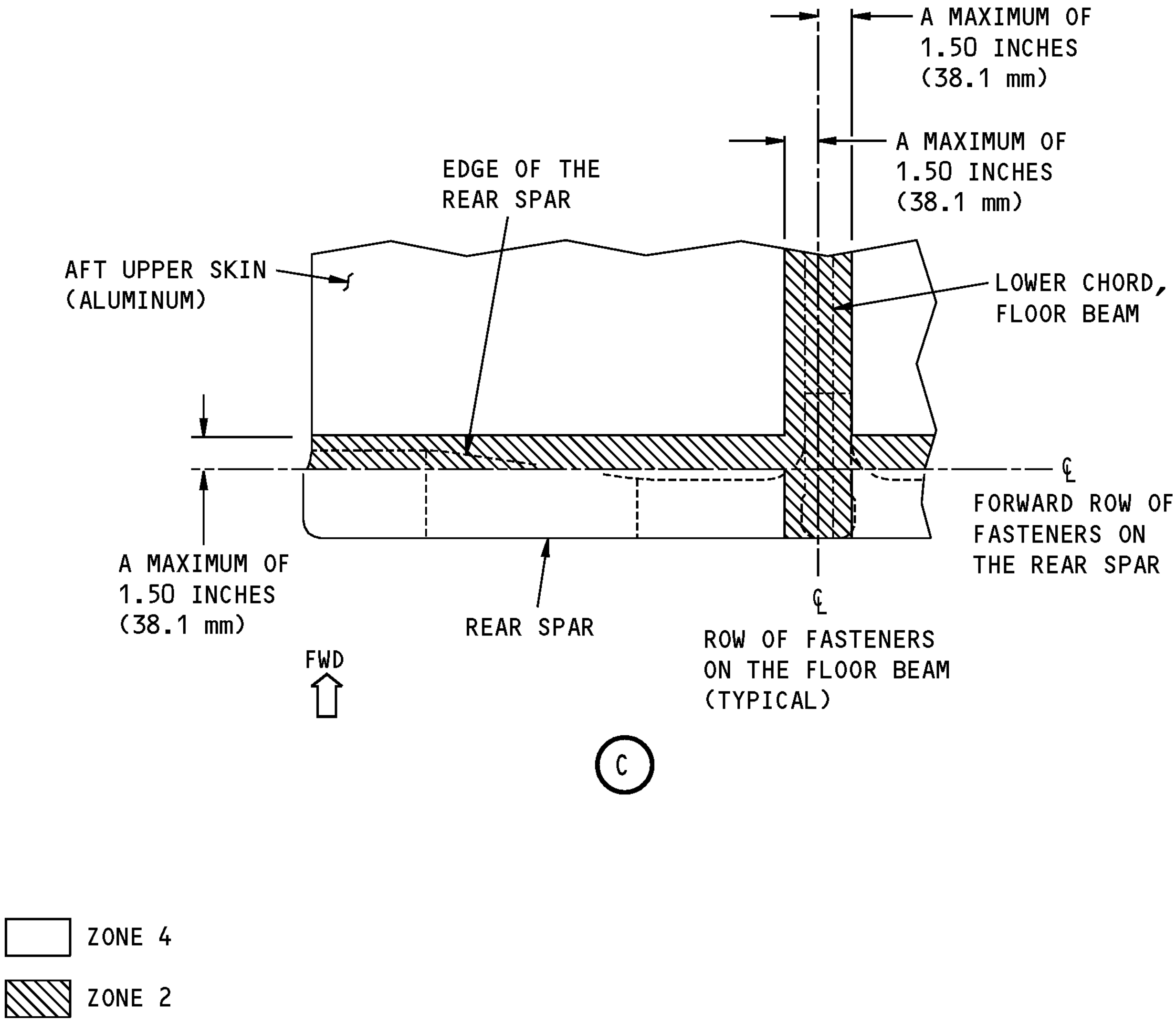

**Wing Center Section Upper Skin Panel Location**
**Figure 101**




**Wing Center Section Allowable Damage Zones**
**Figure 102 (Sheet 1 of 3)**




**Wing Center Section Allowable Damage Zones**
**Figure 102 (Sheet 2 of 3)**




**Wing Center Section Allowable Damage Zones**
**Figure 102 (Sheet 3 of 3)**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| AMM 28-11-00 | Integral Fuel Tanks |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-10-03 | General - Shot Peening Procedures |

## 4. Allowable Damage Limits

A. Zone 1

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A and B if:

1) The maximum decrease in depth is 0.02 inch (0.51 mm)

2) The maximum decrease in cross-sectional area is:

a) 0.020 inch$^2$ (12.90 mm$^2$) between adjacent stringers or between a spar and a stringer

b) 0.20 inch$^2$ (129.03 mm$^2$) between the front spar and the rear spar.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Zone 2

(1) Damage is not permitted.

C. Zone 3

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A and B if:

1) The maximum decrease in depth is 0.03 inch (0.76 mm).

2) The maximum decrease in cross-sectional area is:

a) 0.020 inch$^2$ (12.90 mm$^2$) between adjacent stringers or between a spar and a stringer

b) 0.20 inch$^2$ (129.03 mm$^2$) between the front spar and the rear spar.

D. Zone 4

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

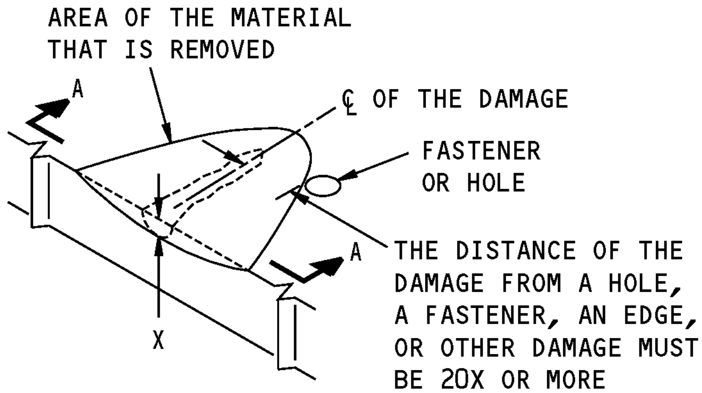
(a) Remove the damage as shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A and B if:

1) The maximum decrease in depth is 0.04 inch (1.02 mm)

2) The maximum decrease in cross-sectional area is:

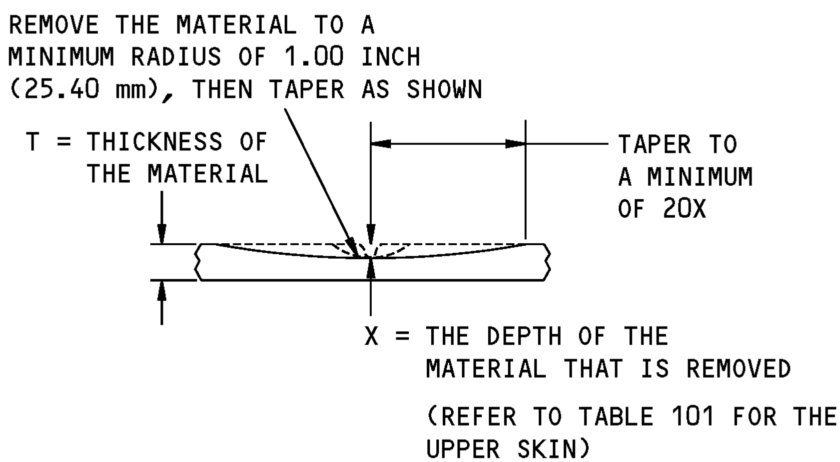
a) 0.020 inch$^2$ (12.90 mm$^2$) between adjacent stringers or between a spar and a stringer

b) 0.20 inch$^2$ (129.03 mm$^2$) between the front spar and the rear spar.



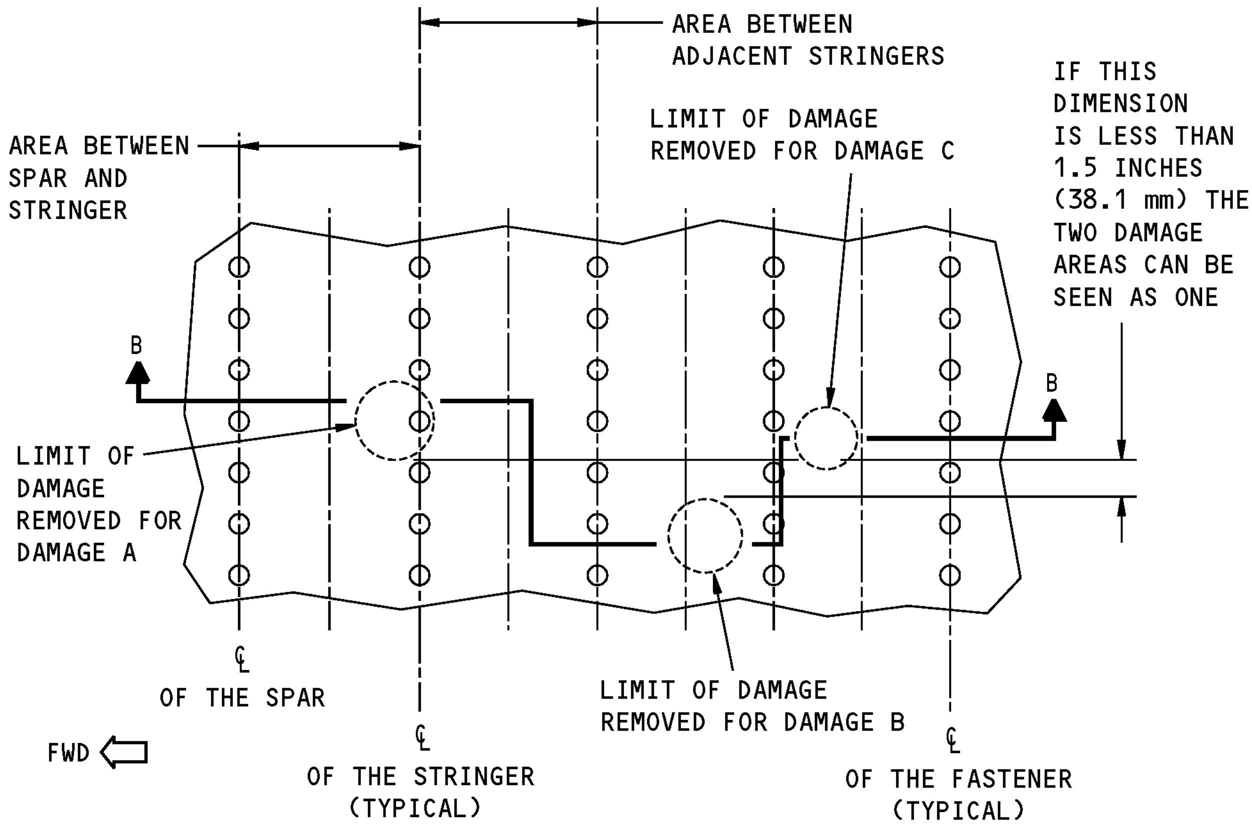

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

A



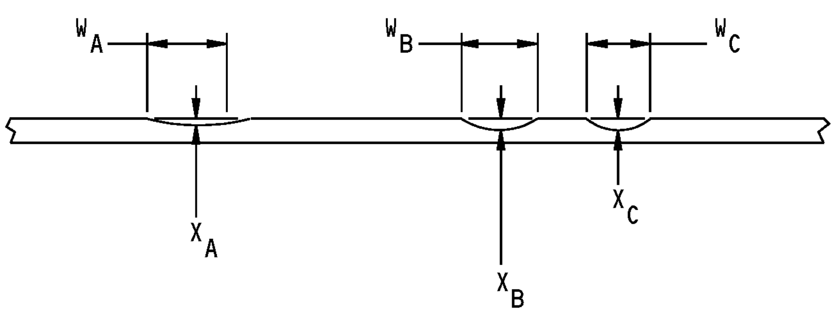

A-A

**Wing Center Section Upper Skin Allowable Damage**
**Figure 103 (Sheet 1 of 2)**




CLEANUP OF DAMAGE THAT IS ALONG A LINE
PERPENDICULAR TO THE STIFFENERS
(REFER TO PARAGRAPH 5 FOR EXAMPLE CALCULATIONS)

B




B–B

**Wing Center Section Upper Skin Allowable Damage**
**Figure 103 (Sheet 2 of 2)**

## 5. Example Calculations

A. Calculate the damage depths and areas shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Detail B to find if the damage is permitted.

**NOTE**: You can make a conservative estimate of the area loss as described below if you can not accurately calculate the area loss. In the example below, the cross sectional area loss is estimated to be 1/2 the width X depth.

(1) Find the allowable damage depth (X) for each damage location shown.

(a) The damage depth at Damage A is 0.015 inch (0.381 mm).

1) This is less than 0.040 inch (0.762 mm) and is permitted.

(b) The damage depth at Damage B is 0.025 inch (0.635 mm).

1) This is less than 0.030 inch (0.762 mm) and is permitted.

(c) The damage depth at Damage C is 0.030 inch (0.762 mm).

1) This is equal to 0.030 inch (0.762 mm) and is permitted.

(2) Calculate the cross-sectional area that has decreased between the stringers (W/2 times X, with W = damage width and X = damage depth). Then find if the decreased area is permitted.

(a) The area that has decreased at Damage A is:

1) 1.6 inches/2 X 0.015 inch = 0.012 inch$^2$ (40.64 mm/2 X 0.381 mm = 7.74 mm$^2$)

a) This is less than 0.020 inch$^2$ (12.90 mm$^2$) and is permitted.

(b) The area that has decreased at Damage B is:

1) 1.5 inches/2 X 0.025 inch = 0.0188 inch$^2$ (38.1 mm/2 X 0.635 mm = 12.09 mm$^2$).

a) This is less than 0.020 inch$^2$ (12.90 mm$^2$) and is permitted.

(c) The area that has decreased at Damage C is:

1) 1.2 inches/2 X 0.030 inch = 0.018 inch$^2$ (30.48 mm/2 X 0.762 mm = 11.61 mm$^2$).

a) This is less than 0.020 inch$^2$ (12.90 mm$^2$) and is permitted.

(3) Calculate the total cross sectional area that has decreased between the front and rear spars. Then find if that decreased area is permitted.

(a) Add the areas that have decreased in Damage A, B, and C as the total area decreased between the front and rear spars.

1) (0.012 + 0.0188 + 0.018) inch$^2$ = 0.0488 inch$^2$ [(7.74 + 12.09 + 11.61) mm$^2$ = 31.44 mm$^2$)

a) This is less than 0.200 square inch (128.85 mm$^2$) and is permitted.

## ALLOWABLE DAMAGE 2 - WING CENTER SECTION LOWER SKIN PANELS

### 1. Applicability

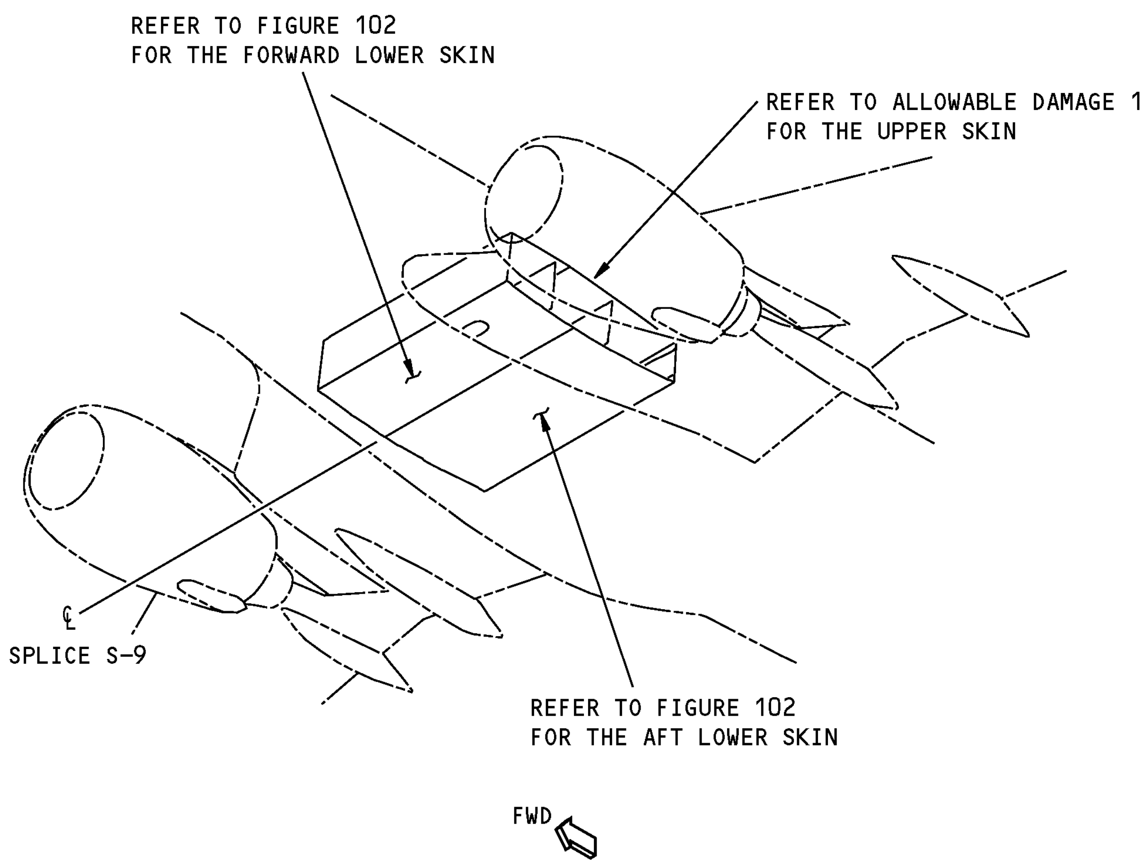
A. This subject gives the allowable damage limits for the lower skin panels of the wing center section shown in Wing Center Section Lower Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.

B. Allowable Damage 2 is applicable to airplanes that have not had winglets installed

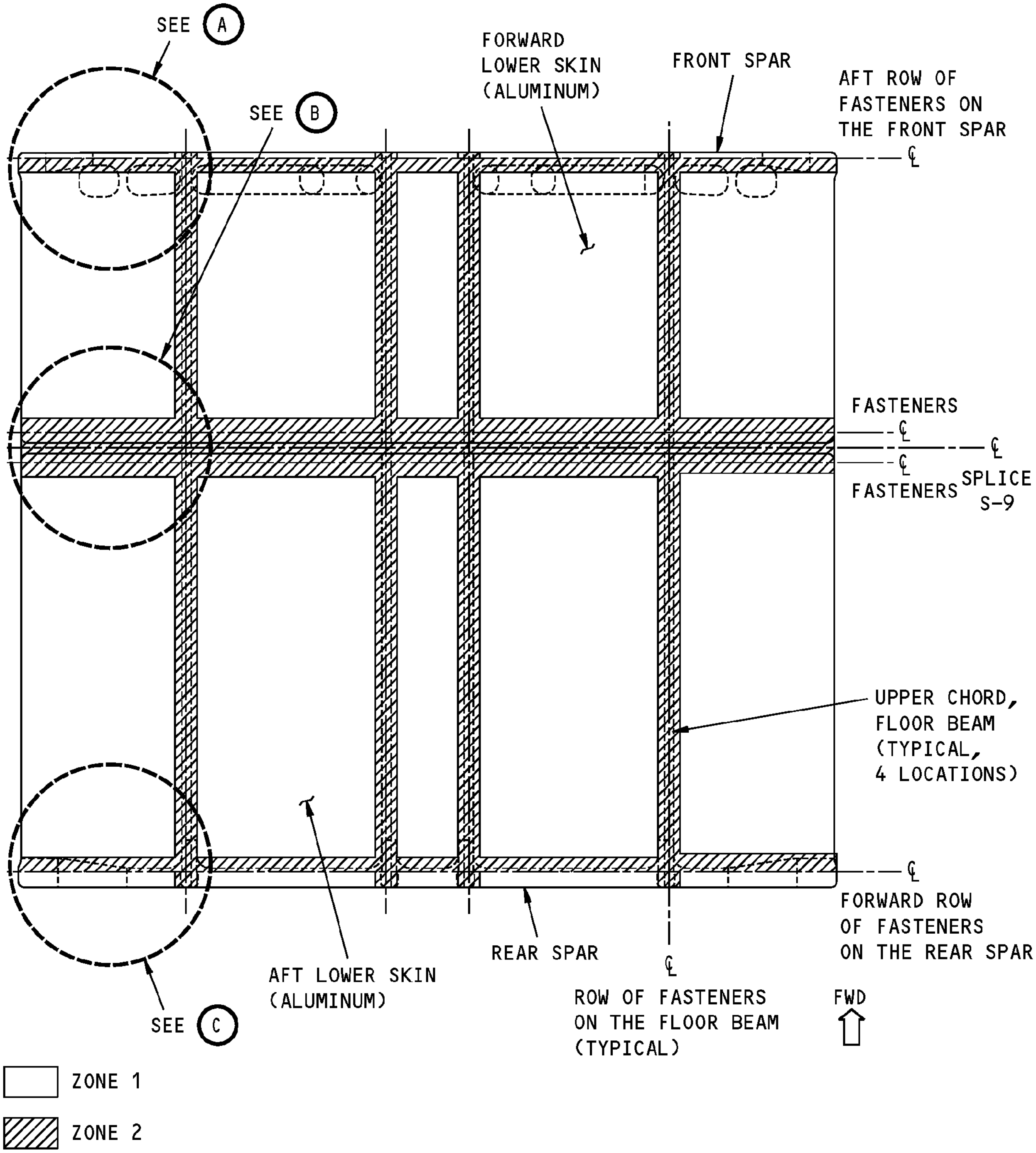
C. Allowable Damage 2 is not applicable to:

(1) Airplanes that have had winglets installed as part of the production of the airplane. Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets.

(2) Airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

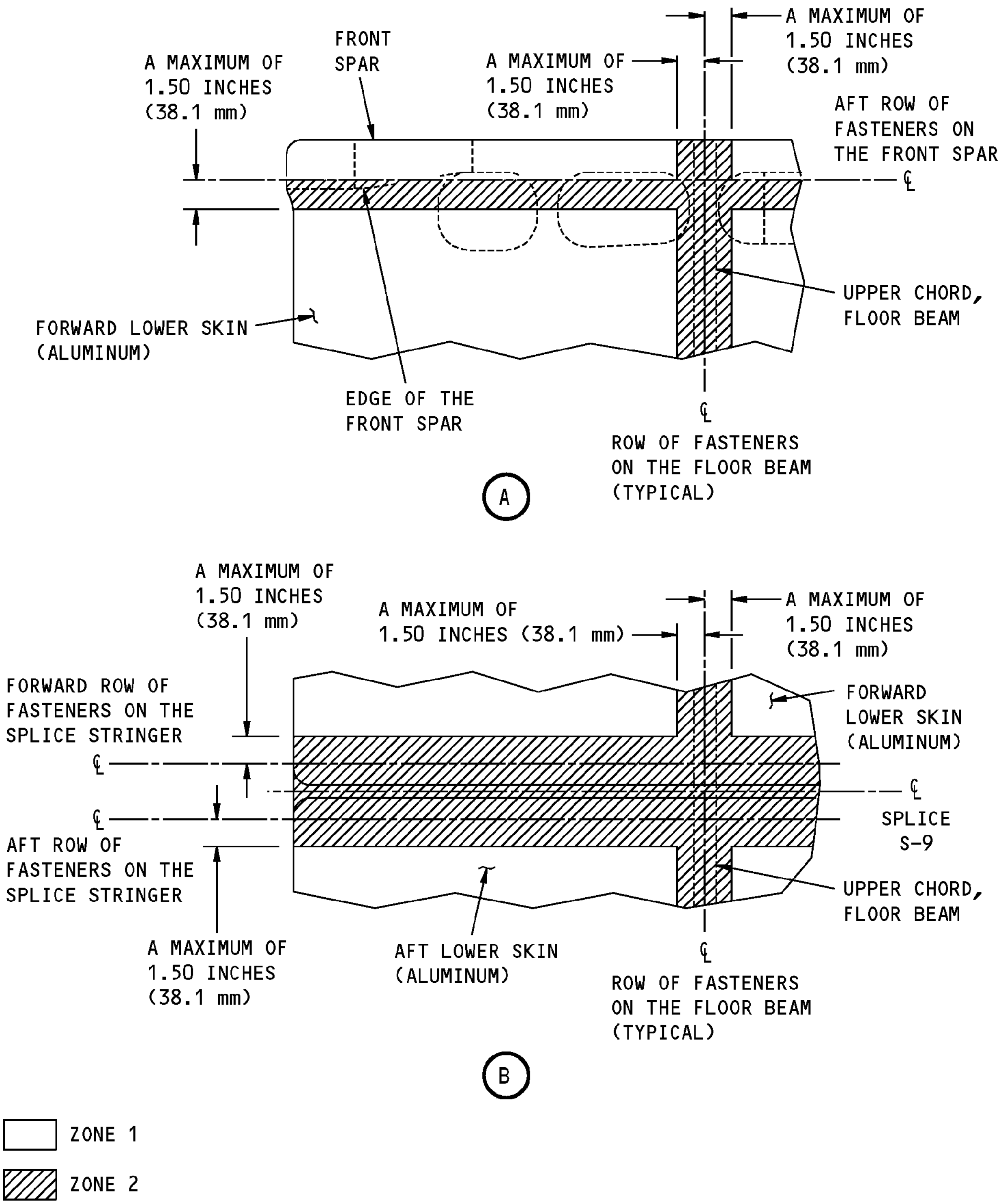



**Wing Center Section Lower Skin Panel Location**
**Figure 101**

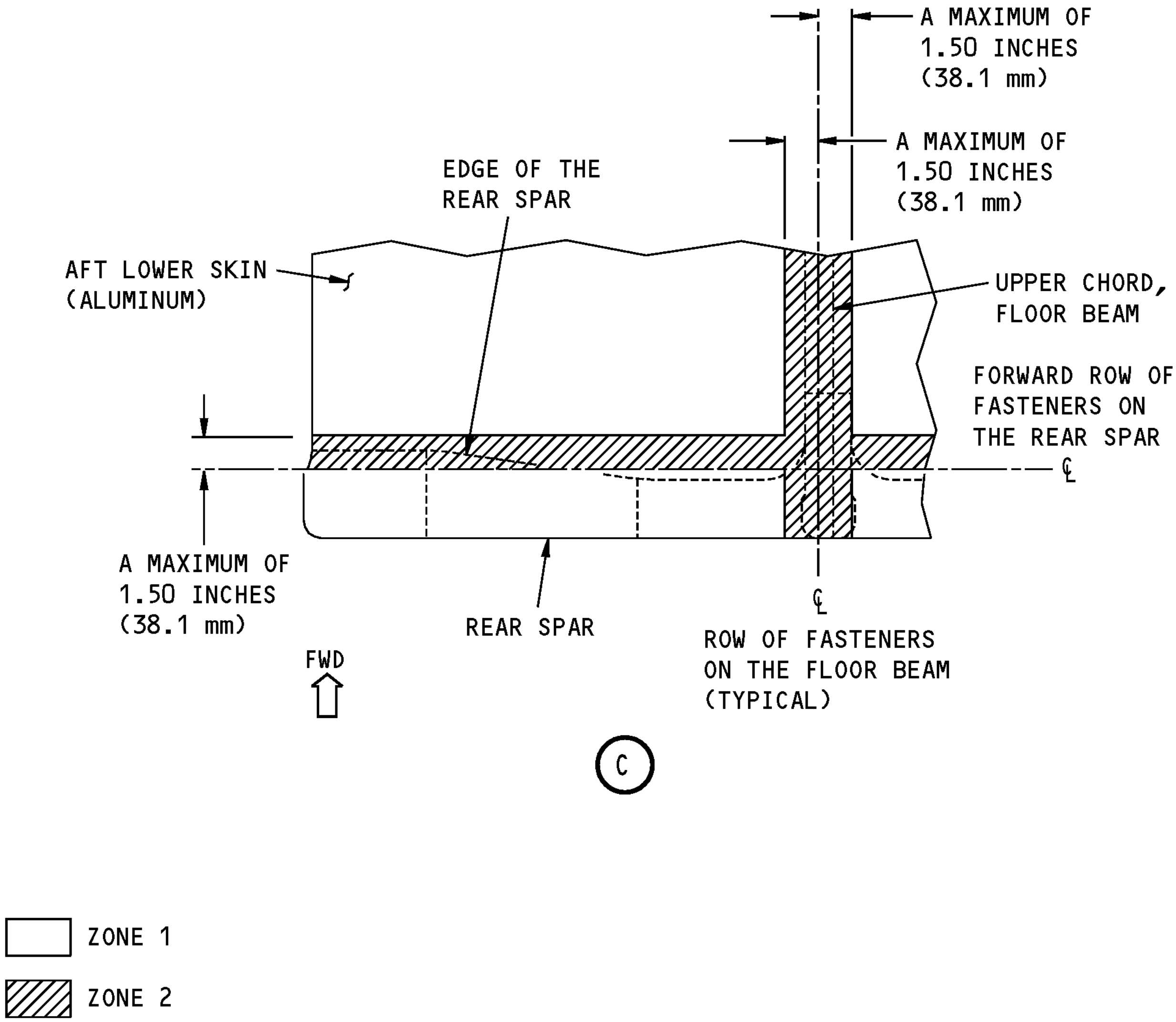



BOTTOM VIEW, LOWER SKINS - WING CENTER SECTION

**Wing Center Section Allowable Damage Zones**
**Figure 102 (Sheet 1 of 3)**




**Wing Center Section Allowable Damage Zones**
**Figure 102 (Sheet 2 of 3)**




**Wing Center Section Allowable Damage Zones**
**Figure 102 (Sheet 3 of 3)**

**2. General**

A. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasive materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do as follows:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the lower skin panels.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(3) Apply a layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM 51-21-00/701.

**3. References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

**4. Allowable Damage Limits**

A. Zone 1

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Lower Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A and B .

1) Refer to Table 101/ALLOWABLE DAMAGE 2 for the maximum depth, maximum loss of area, and maximum total loss of area.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

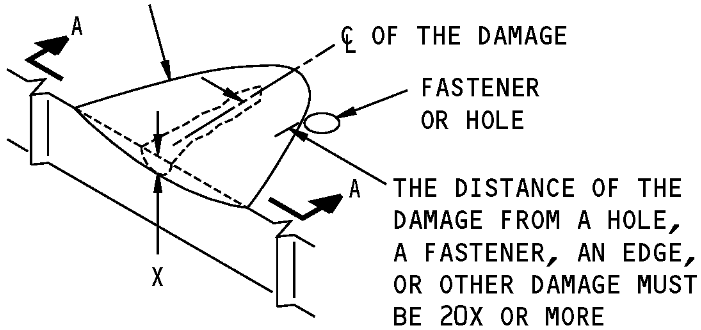
B. Zone 2

(1) Damage is not permitted.

**Table 101**:

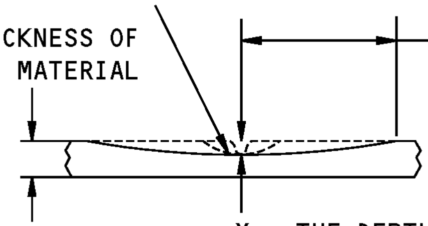
| ZONE 1 | MAXIMUM DEPTH OF THE DAMAGE AFTER CLEANUP IN INCHES | MAXIMUM LOSS OF CROSS SECTIONAL AREA BETWEEN ADJACENT STRINGERS OR BETWEEN A SPAR AND A STRINGER IN INCHES$^2$ | TOTAL MAXIMUM LOSS OF CROSS SECTIONAL AREA BETWEEN THE FRONT SPAR AND THE REAR SPAR IN INCHES$^2$ |
|---|---|---|---|
| FORWARD LOWER SKIN (FRONT SPAR TO S-9) | 0.025 | 0.040 | 0.400 |
| AFT LOWER SKIN (S-9 TO S-5) | 0.035 | 0.060 | 0.400 |
| AFT LOWER SKIN (S-5 TO REAR SPAR) | 0.046 | 0.065 | 0.400 |

**NOTE**: Allowable damage limits are not applicable if there is damage to a stiffener which is not repaired in the same area.



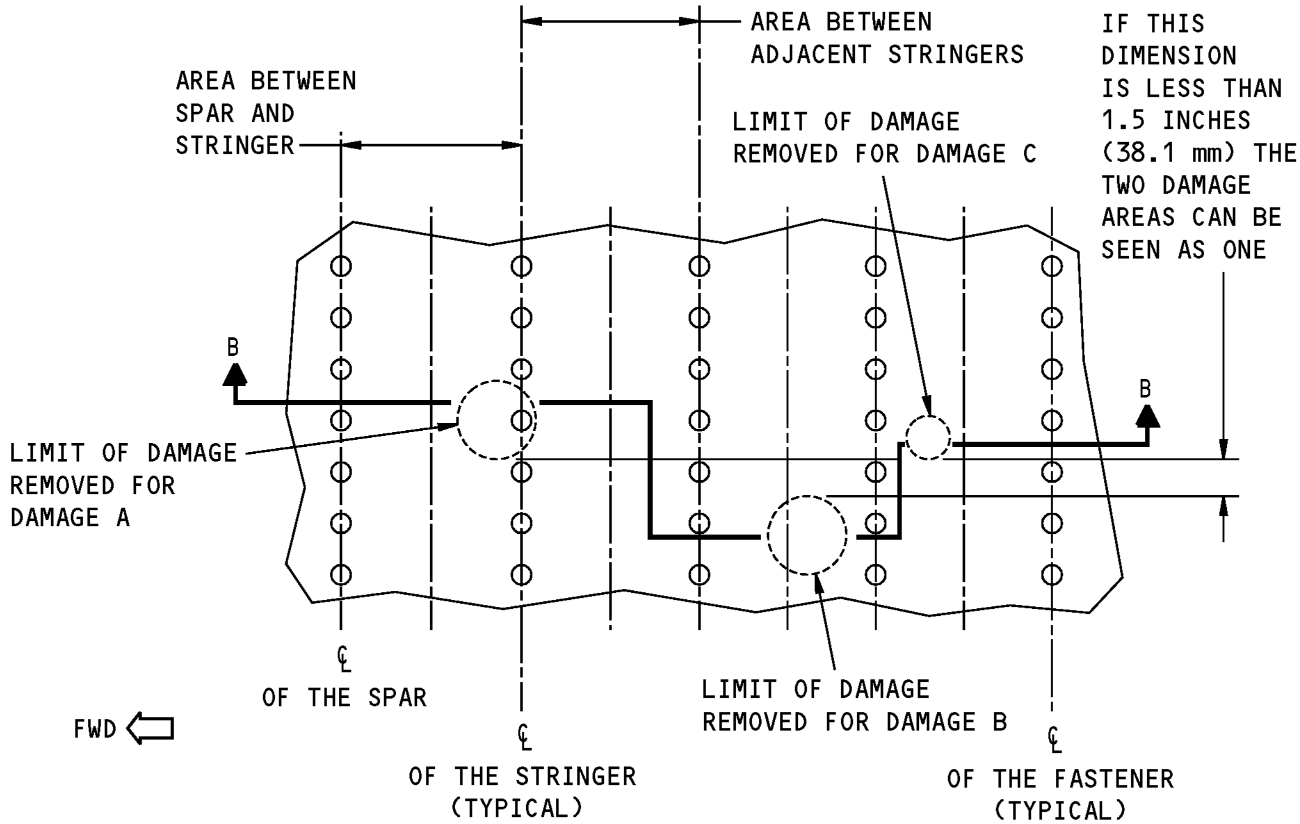

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

A



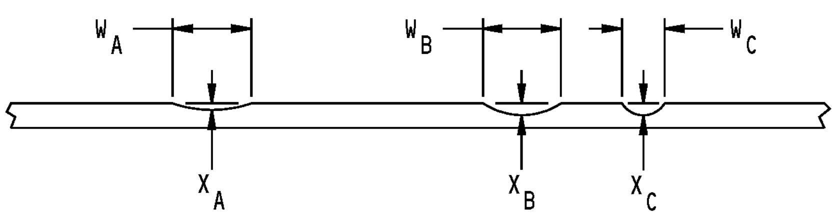

A-A

**Wing Center Section Lower Skin Allowable Damage**
**Figure 103 (Sheet 1 of 2)**




CLEANUP OF DAMAGE THAT IS ALONG A LINE PERPENDICULAR TO THE STIFFENERS (REFER TO PARAGRAPH 5 FOR EXAMPLE CALCULATIONS)

B




B-B

**Wing Center Section Lower Skin Allowable Damage**
**Figure 103 (Sheet 2 of 2)**

5. **Example Calculations**

A. Calculate the damage depths and areas shown in Wing Center Section Lower Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail B . Refer Table 101/ALLOWABLE DAMAGE 2 to find if the damage is permitted.

**NOTE**: For the examples that follow, the calculations will be shown for damage in Zone 1 between the front spar and Stringer S-9. You can make a conservative estimate of the area as described below if you can not accurately calculate the area loss. In the example below, the cross sectional area loss is estimated to be 1/2 the width X depth.

(1) Find the allowable damage depth (X) for each damage location shown.

(a) The damage depth at Damage A is 0.015 inch.

1) This is less than 0.025 inch and is permitted.

(b) The damage depth at Damage B is 0.020 inch.

1) This is less than 0.025 inch and is permitted.

(c) The damage depth at Damage C is 0.025 inch.

1) This is equal to 0.025 inch and is permitted.

(2) Calculate the cross-sectional area that has decreased between the stringers (W/2 times X, with W = damage width and X = damage depth). Then find if the decreased area is permitted.

(a) The area that has decreased at Damage A is:

1) 1.6 inches/2 X 0.015 inch = 0.012 inch$^2$.

a) This is less than 0.040 inch$^2$ and is permitted.

(b) The area that has decreased at Damage B is:

1) 2.5 inches/2 X 0.020 inch = 0.025 inch$^2$.

a) This is less than 0.040 inch$^2$ and is permitted.

(c) The area that has decreased at Damage C is:

1) 3.0 inches/2 X 0.025 inch = 0.037 inch$^2$.

a) This is less than 0.040 inch$^2$ and is permitted.

(3) Calculate the total cross-sectional area that has decreased between the front and rear spars. Then find if that decreased area is permitted.

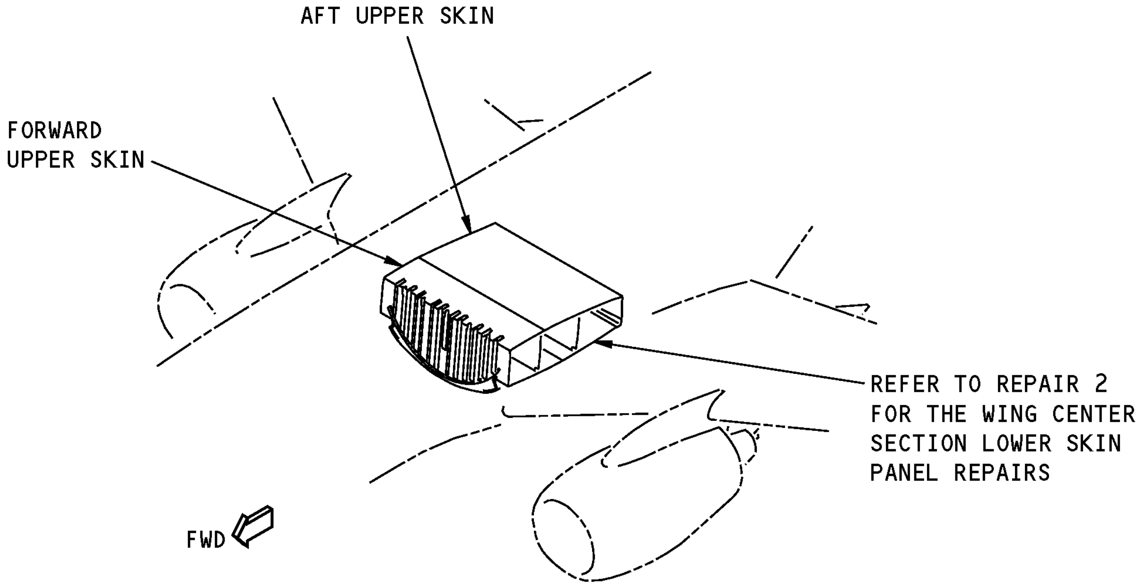
(a) Add the areas that have decreased in Damage A, B, and C as the total area decreased between the front and rear spars.

(b) Add the areas that have decreased in Damage A, B, and C as the total area decreased between the front and rear spars.

1) [0.012 + 0.025 + 0.037] inch$^2$ = 0.074 inch$^2$

2) This is less than 0.400 inch$^2$ and is permitted.

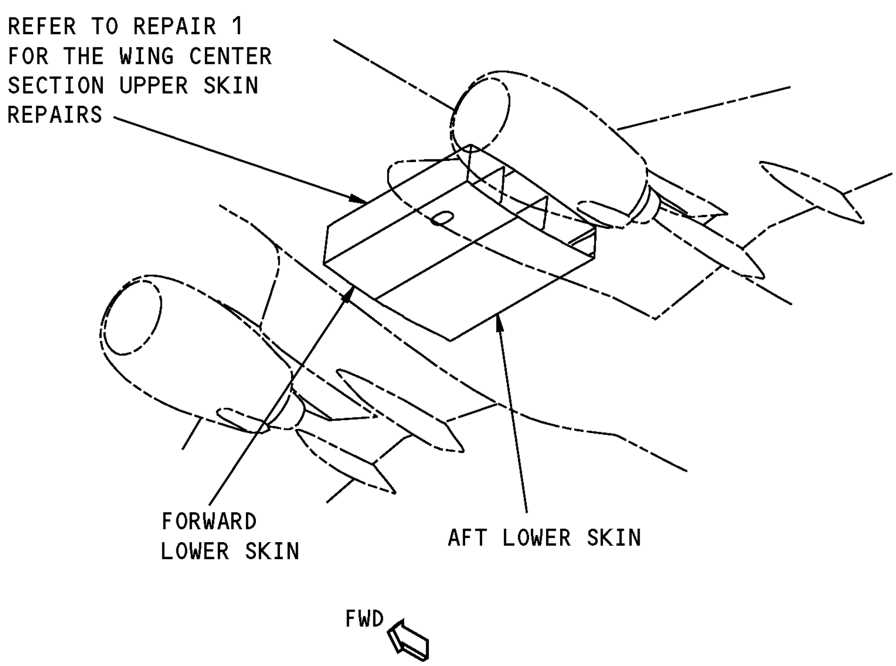
## REPAIR 1 - WING CENTER SECTION UPPER SKIN PANELS




NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

**Wing Center Section Upper Skin Panel Repairs**
**Figure 201**

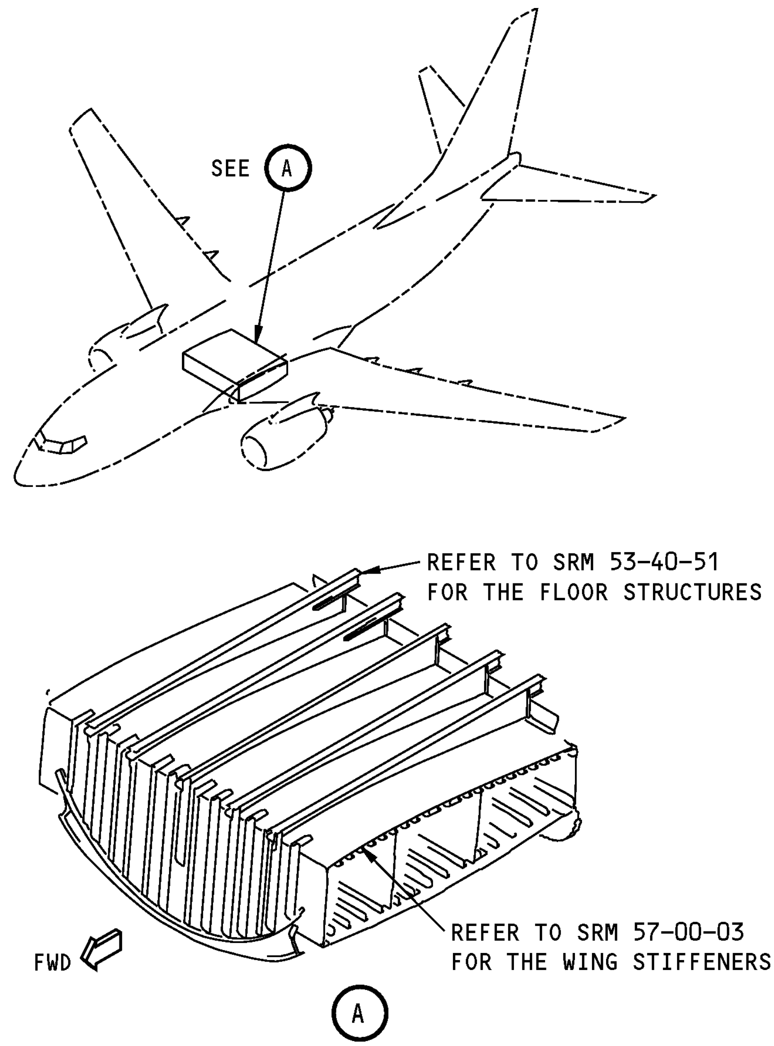
## REPAIR 2 - WING CENTER SECTION LOWER SKIN PANELS




NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

**Wing Center Section Lower Skin Panel Repairs**
**Figure 201**

## ALLOWABLE DAMAGE 1 - WING CENTER SECTION STIFFENERS




**Wing Center Section Stiffener Allowable Damage**
**Figure 101**

## REPAIR 1 - UPPER ZEE STRINGER ON THE CENTER WING

### 1. Applicability

A. Repair 1 is applicable for damage to an upper zee stringer on the center wing. A typical zee stringer is shown in Typical Zee Stringer Section, Figure 201/REPAIR 1.

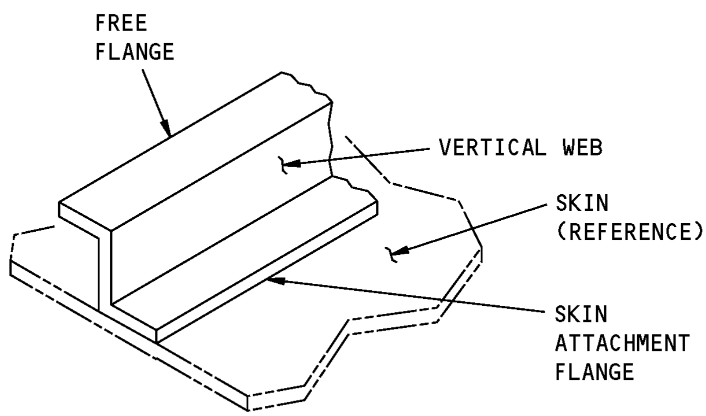
B. Repair 1 is not applicable to upper zee stringers that have:

(1) A free flange that is thicker than 0.140 inch

(2) A skin attachment flange that is thicker than 0.180 inch.

### 2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.




**Typical Zee Stringer Section**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |

(Continued)

| Reference | Title |
|---|---|
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |

4. **Repair Instructions**

**WARNING**: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION**: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

**CAUTION**: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 1. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 1.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | 2 | Use 7075-O material, 0.050 inch in thickness. Heat treat to T6 condition after you form the part |
| [2] | Angle | 2 | Use 7075-O material, 0.050 inch in thickness. Heat treat to T6 condition after you form the part |
| [3] | Strap | 1 | Use 7075-T6 material, 0.080 inch in thickness |
| [4] | Strap | 1 | Use 7075-T6 material, 0.125 inch in thickness |

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [5] | Filler | 1 | Make the part from the same type of extrusion as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material. |

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [5] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

(5) Apply a layer of BMS 5-81, Type II secondary fuel barrier sealant to the upper surface of the upper wing skin.

(a) Apply the BMS 5-81 sealant to the heads of the initial fasteners that you replaced when you installed the repair.

M. Reinstall the fuel tank access doors as necessary.

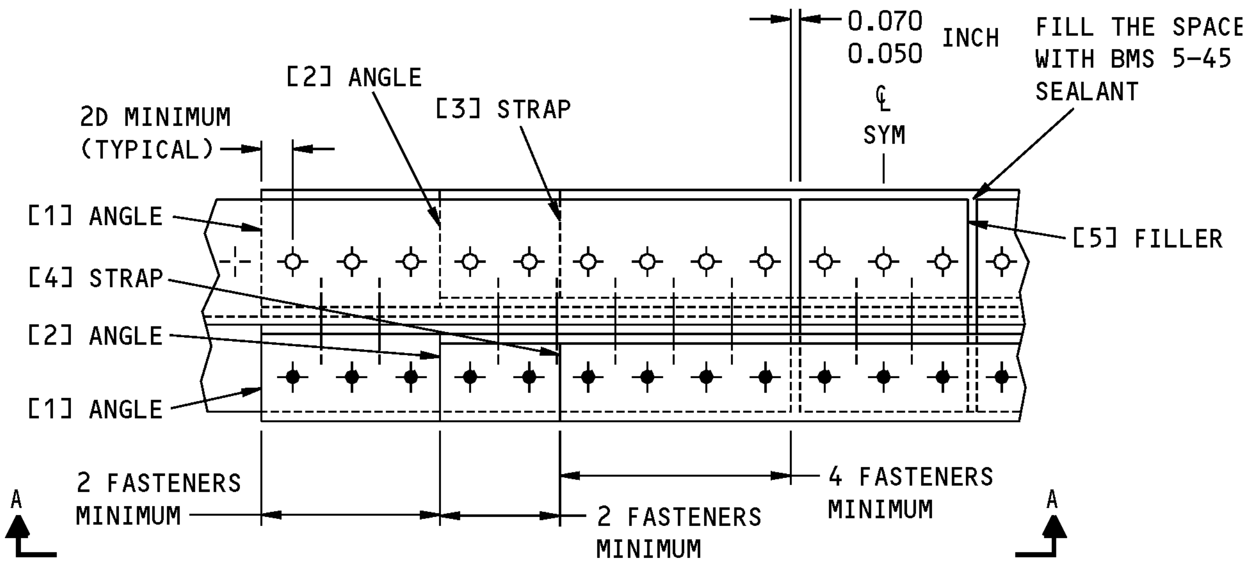
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

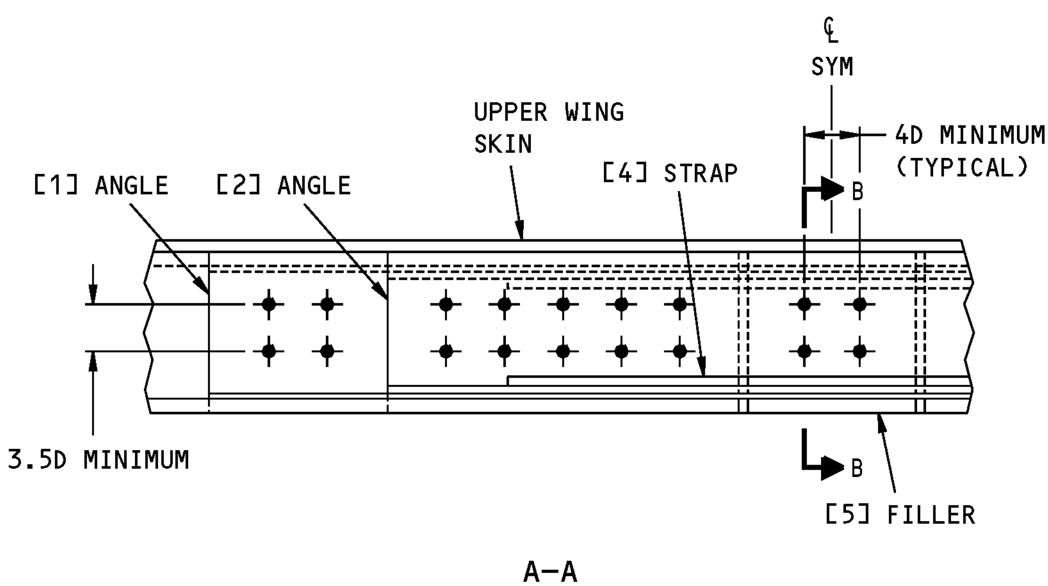
N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

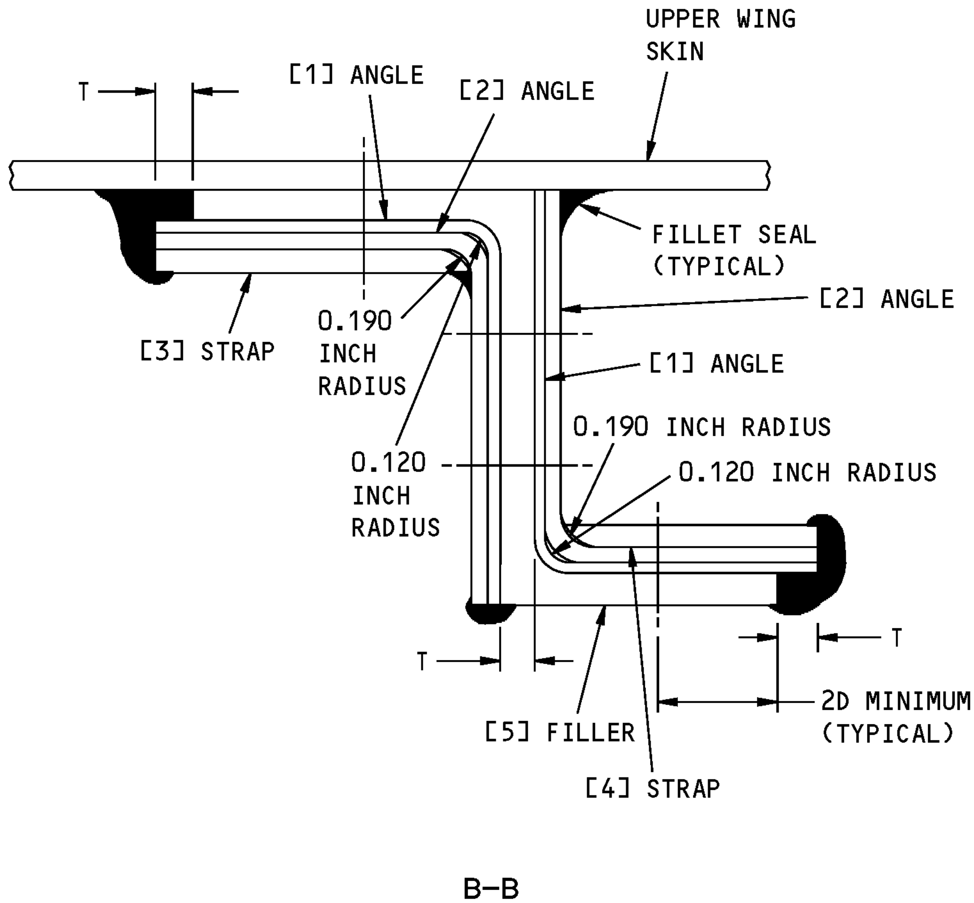
(2) Do a leak test to make sure there are no leaks in the fuel tank.




NOTE: THE UPPER WING SKIN IS NOT SHOWN.




**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

-ф- INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30R8 COLLAR.

-✦- REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30M8 COLLAR.

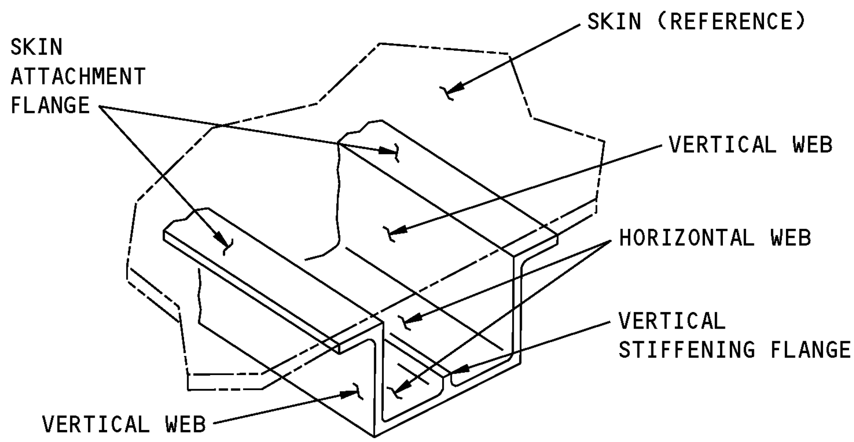
**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

## REPAIR 2 - UPPER VENT STRINGER ON THE CENTER WING ADJACENT TO A PADUP AREA

1. **Applicability**

   A. Repair 2 is applicable to damage an upper vent stringer on the center wing that is adjacent to a padup area. A typical upper vent stringer is shown in Typical Vent Stringer Section, Figure 201/REPAIR 2.

   B. Repair 2 is applicable at locations where there is not sufficient clearance between the repair parts and the stringer padup area.




**Typical Vent Stringer Section**
**Figure 201**

2. **General**

   A. Repair 2 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

   B. The repair that follows is an alternative to Repair 2:

      (1) Repair 3 is for damage to an upper vent stringer on the center wing that is not adjacent to a padup area. Use this repair when there is sufficient clearance between the repair parts and the stringer padup area.

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |

(Continued)

| Reference | Title |
|---|---|
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |

## 4. Repair Instructions

**WARNING:** MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION:** MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

**CAUTION:** BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 2. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 2.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | As Necessary | Use 7075-O material with a thickness that is the same as the depth of the recess. Heat treat to T6 condition after you form the part. If there is not a recess, do not use the part |
| [2] | Angle | 2 | Use 7075-O material, 0.063 inch in thickness. Heat treat to the T6 condition after you form the part |
| [3] | Angle | 2 | Use 7075-O material, 0.080 inch in thickness. Heat treat to the T6 condition after you form the part |
| [4] | Plate | 1 | Use 7075-T6 material, 0.050 inch in thickness |
| [5] | Plate | 1 | Use 7075-T6 material, 0.063 inch in thickness |
| [6] | Plate | 1 | Use 7075-T6 material, 0.080 inch in thickness |
| [7] | Stringer | 1 | Make the part from the same type of extrusion as the initial stringer. Use 7055-T77511 material. As an al- ternative, you can machine the part from 7075-T6 material |

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] stringer and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 2.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

(5) Apply a layer of BMS 5-81, Type II secondary fuel barrier sealant to the upper surface of the upper wing skin.

(a) Apply the BMS 5-81 sealant to the heads of the initial fasteners that you replaced when you installed the repair.

M. Reinstall the fuel tank access doors as necessary.

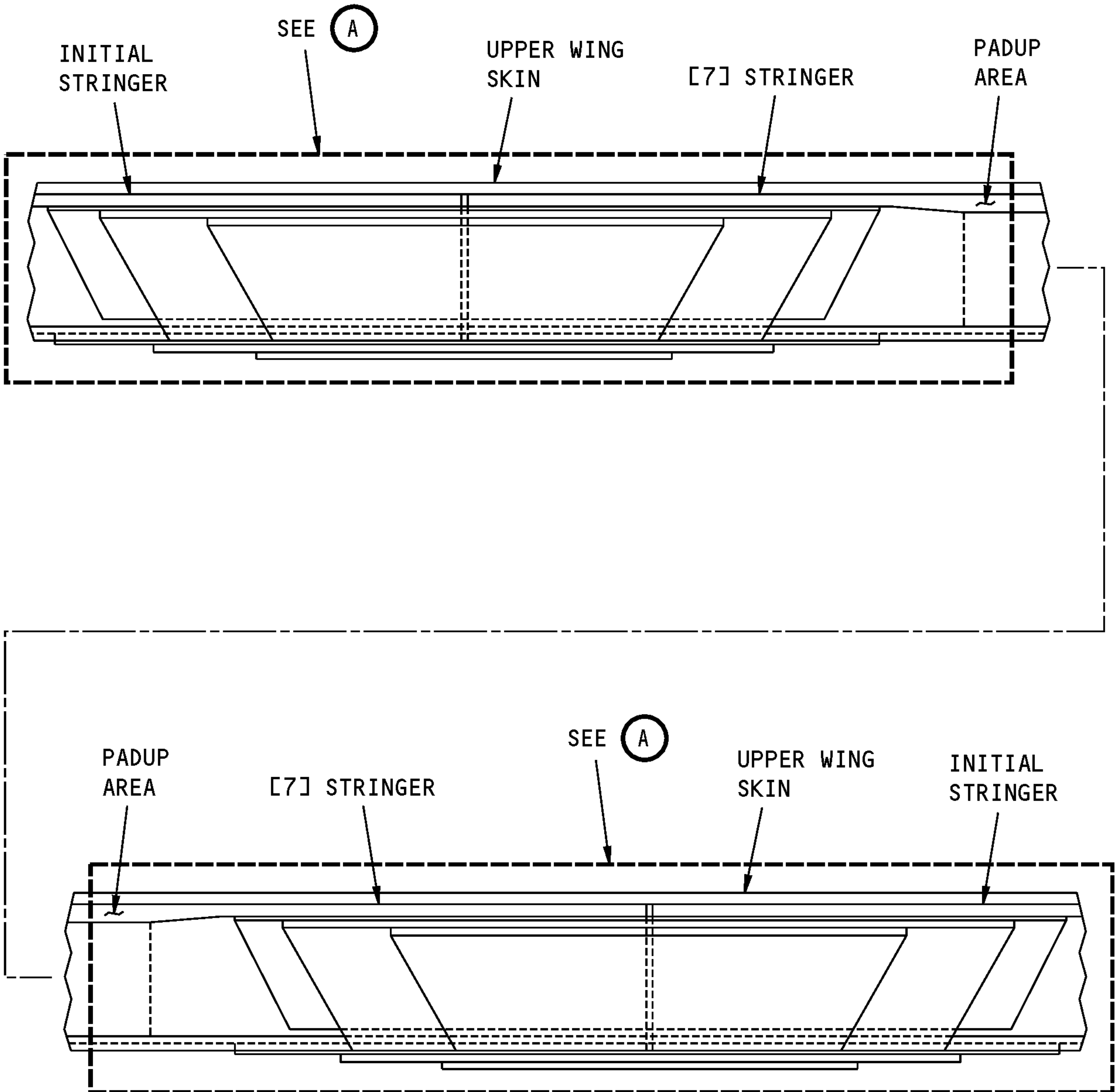
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

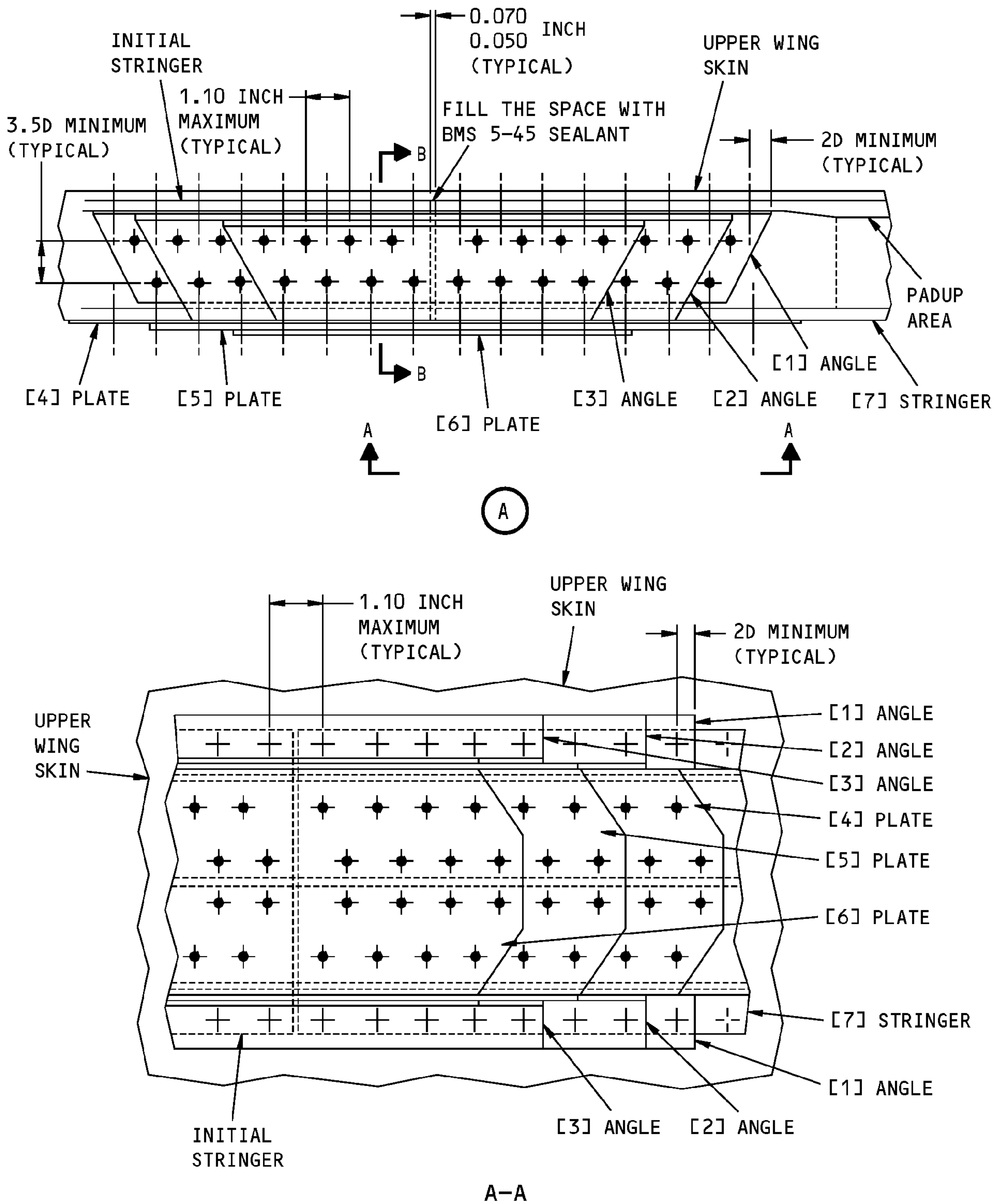
N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

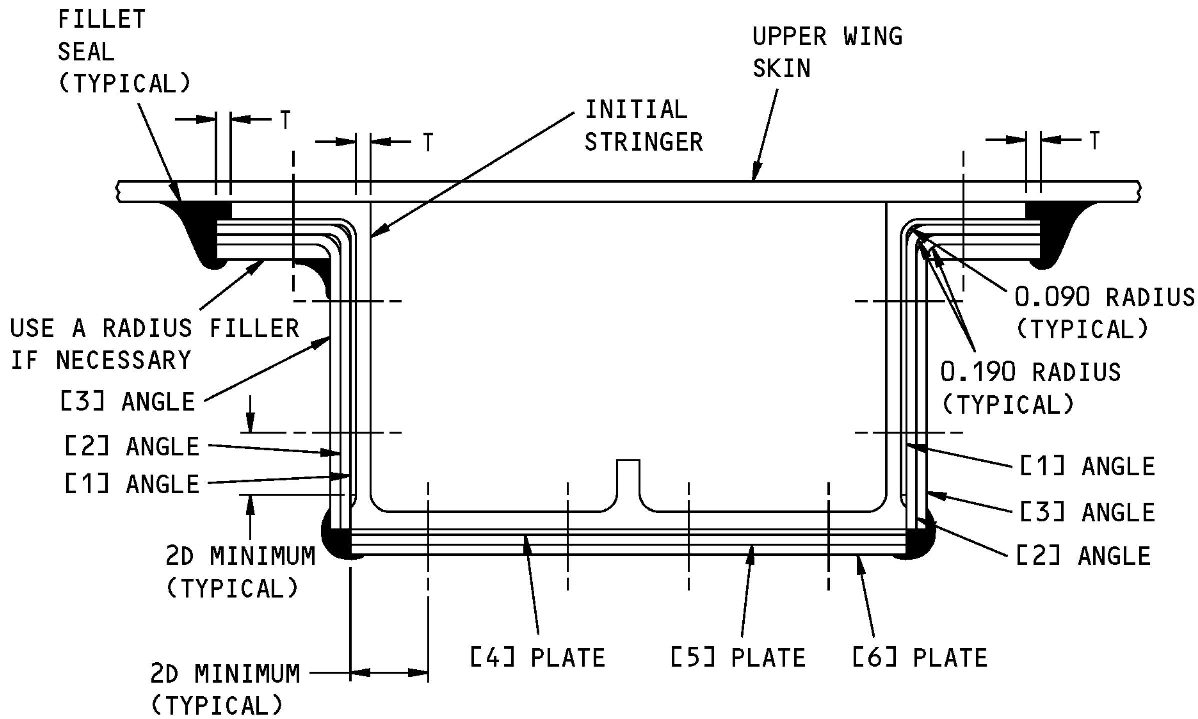
(2) Do a leak test to make sure there are no leaks in the fuel tank.




**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 3)**




**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 3)**




B-B

FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR.

✦ REPAIR FASTENER LOCATION. INSTALL A MS21141-08()P BLIND BOLT.

**Layout of the Repair Parts**
**Figure 202 (Sheet 3 of 3)**

## REPAIR 3 - UPPER VENT STRINGER ON THE CENTER WING NOT ADJACENT TO A PADUP AREA

1. **Applicability**

   A. Repair 3 is applicable for damage to a vent stringer on the center wing that is not adjacent to padup area. A typical vent stringer is shown in Typical Vent Stringer Section, Figure 201/REPAIR 3.

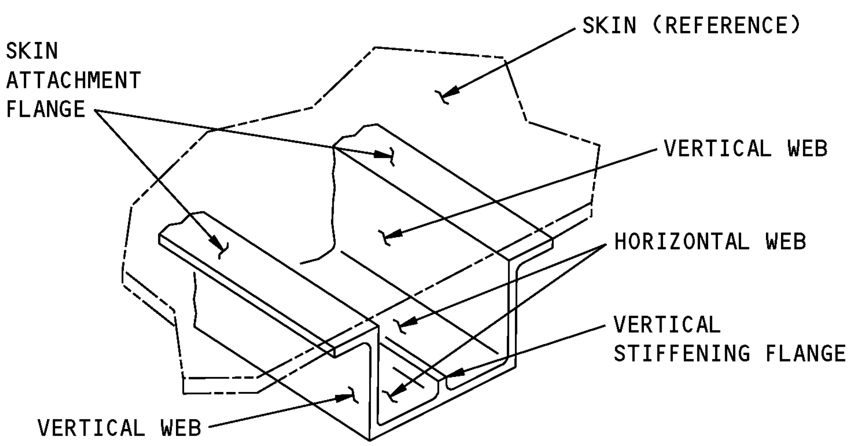
   B. Repair 3 is applicable at locations where there is sufficient clearance between the repair parts and the stringer padup area.

2. **General**

   A. Repair 3 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

   B. The repair that follows is an alternative to Repair 3:

      (1) Repair 2 is a repair for damage to a upper vent stringer on the center wing that is adjacent to a padup area. Use this repair when there is not sufficient clearance between the repair parts and the stringer padup.




**Typical Vent Stringer Section**
**Figure 201**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |

(Continued)

| Reference | Title |
|---|---|
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| 57-10-01, REPAIR 2 | Wing Center Section Lower Skin Panels |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |

## 4. Repair Instructions

**WARNING**: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION**: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

**CAUTION**: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 3. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 3.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | As Necessary | Use 7075-O material that is the same thickness as the depth of the recess. Heat treat to T6 condition after you form the part. If there is not a recess, do not use the part |
| [2] | Angle | 2 | Use 7075-O material, 0.063 inch in thickness. Heat treat to T6 condition after you form the part |
| [3] | Angle | 2 | Use 7075-O material, 0.080 inch in thickness. Heat treat to T6 condition after you form the part |
| [4] | Plate | 1 | Use 7075-T6 material, 0.050 inch in thickness |
| [5] | Plate | 1 | Use 7075-T6 material, 0.063 inch in thickness |
| [6] | Plate | 1 | Use 7075-T6 material, 0.080 inch in thickness |
| [7] | Filler | 1 | Make the part from the same type of extrusion as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material |

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 3.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

(5) Apply a layer of BMS 5-81, Type II secondary fuel barrier sealant to the upper surface of the upper wing skin.

(a) Apply the BMS 5-81 sealant to the heads of the initial fasteners that you replaced when you installed the repair.

M. Reinstall the fuel tank access doors as necessary.

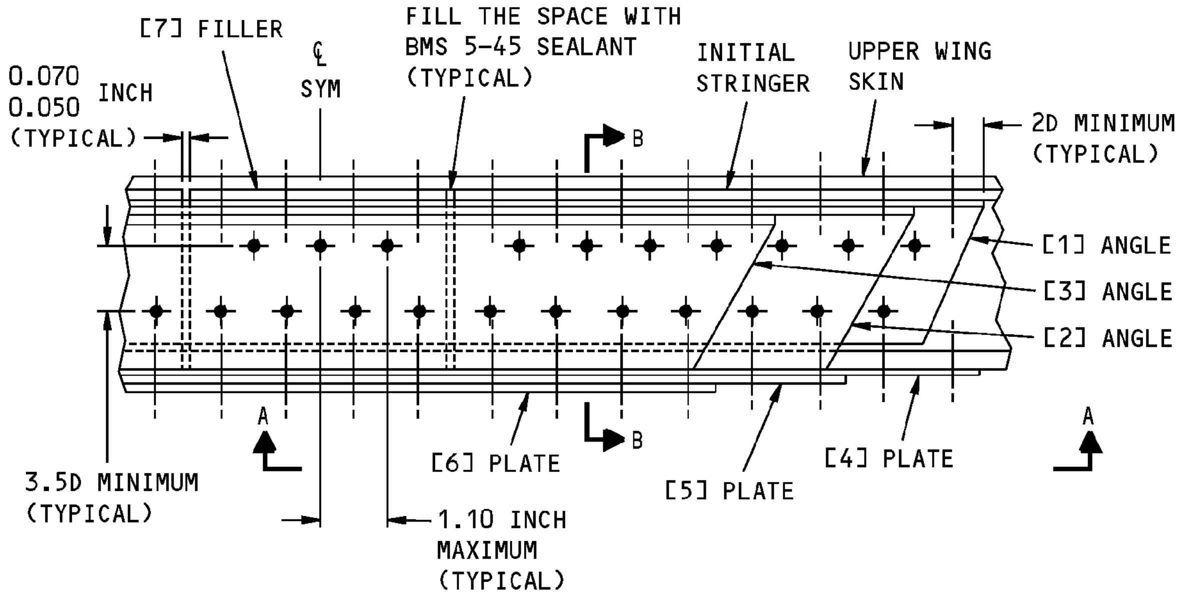
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

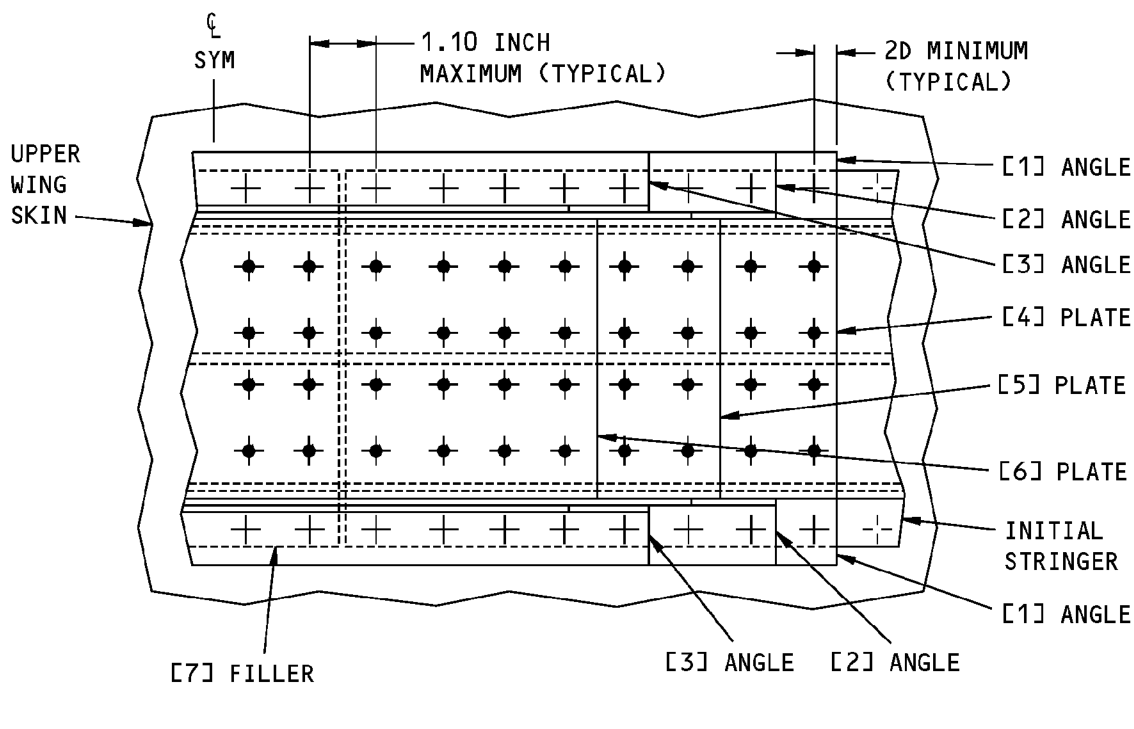
(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

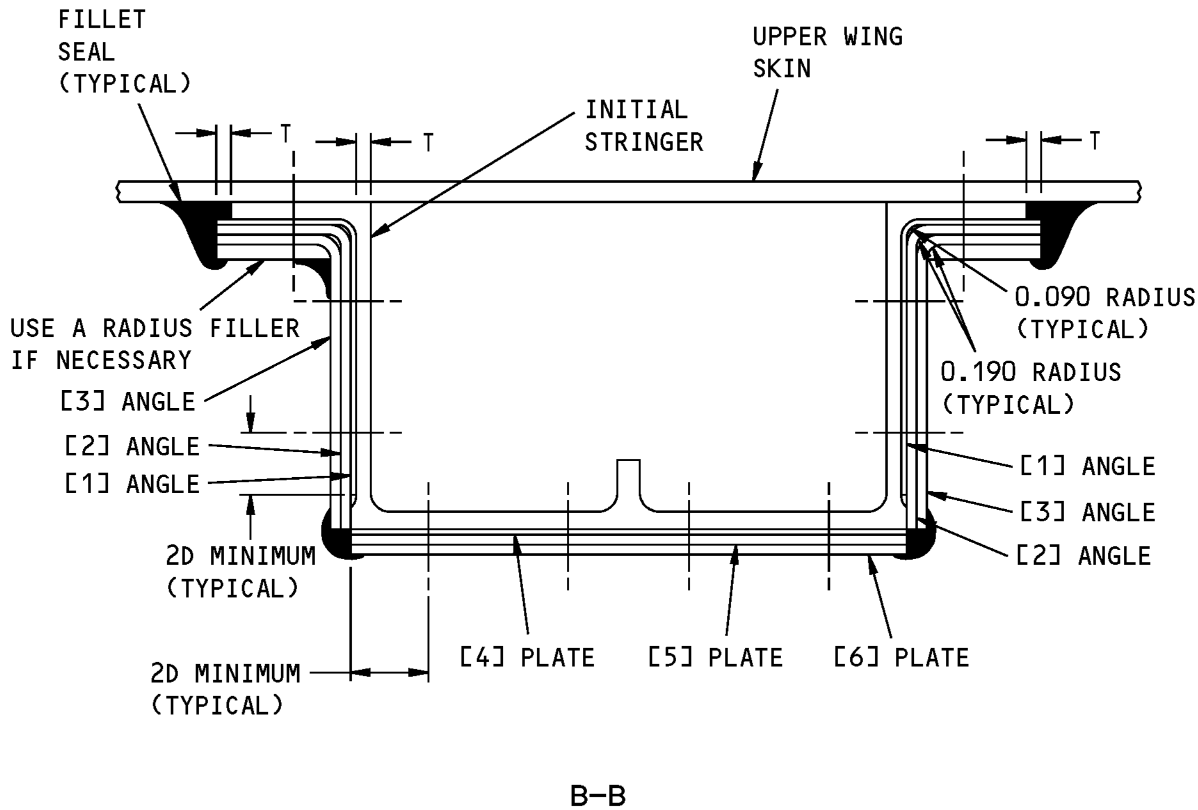
(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.




Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC COLLAR.

⌖ REPAIR FASTENER LOCATION. INSTALL A MS21141-08()P BLIND BOLT.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

## REPAIR 4 - LOWER SPLICE STRINGER ON THE CENTER WING ADJACENT TO A PADUP

1. **Applicability**

   A. Repair 4 is applicable for damage to a lower splice stringer on the center wing adjacent to a padup. A typical splice stringer is shown in Typical Splice Stringer Section, Figure 201/REPAIR 4.

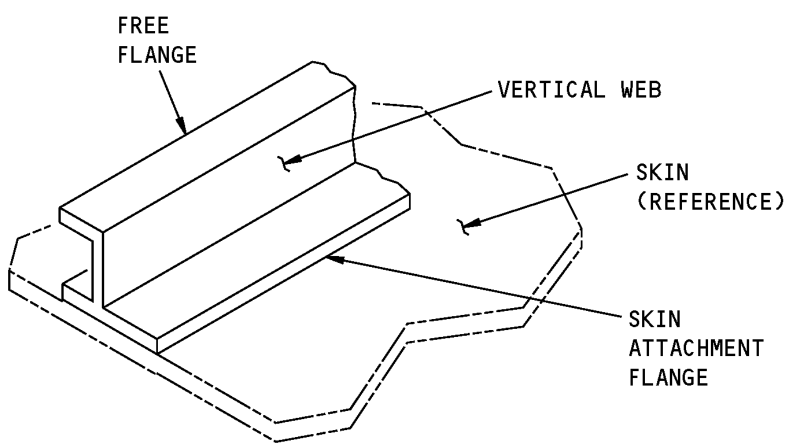
   B. Repair 4 is applicable at locations where there is not sufficient clearance between the repair parts and the stringer padup area.

2. **General**

   A. Repair 4 is a Category B repair. Refer to 51-00-06 for the definitions of the different classes of repairs. Refer to the inspection instructions given in Paragraph 5./REPAIR 4

   B. The repair that follows is an alternative to Repair 4:

      (1) Repair 5 is a repair for damage to a lower splice stringer on the center wing that is not adjacent to a padup area. Use this repair when there is sufficient clearance between the repair parts and the stringer padup.




**Typical Splice Stringer Section**
**Figure 201**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |

(Continued)

| Reference | Title |
|---|---|
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-09 | COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 8 | Inspection of Subsurface Cracks in Aluminum Structure |
| 737 NDT Part 6, 51-00-00, Figure 9 | Inspection of Subsurface Cracks at Fastener Holes in Aluminum Structure |

## 4. Repair Instructions

**WARNING**: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION**: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

**CAUTION**: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 4. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 4.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | 2 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [2] | Channel | 2 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [3] | Angle | 2 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [4] | Channel | 2 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [5] | Plate | 2 | Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part |
| [6] | Plate | 2 | Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part |
| [7] | Stringer | 1 | Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy. As an alternative, you can machine the part from 2024-T42 aluminum alloy material |

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] stringer and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 4.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.

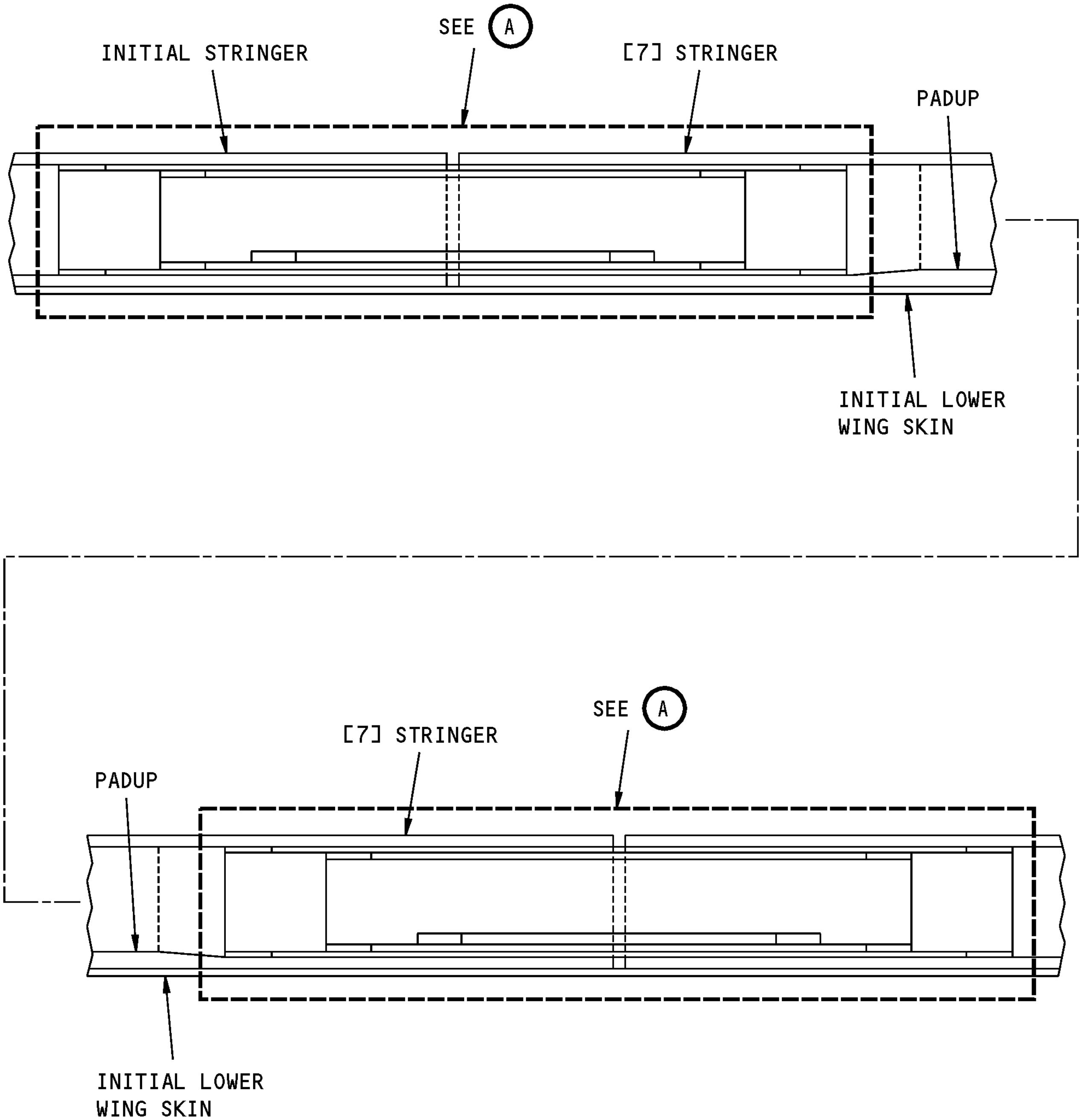
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

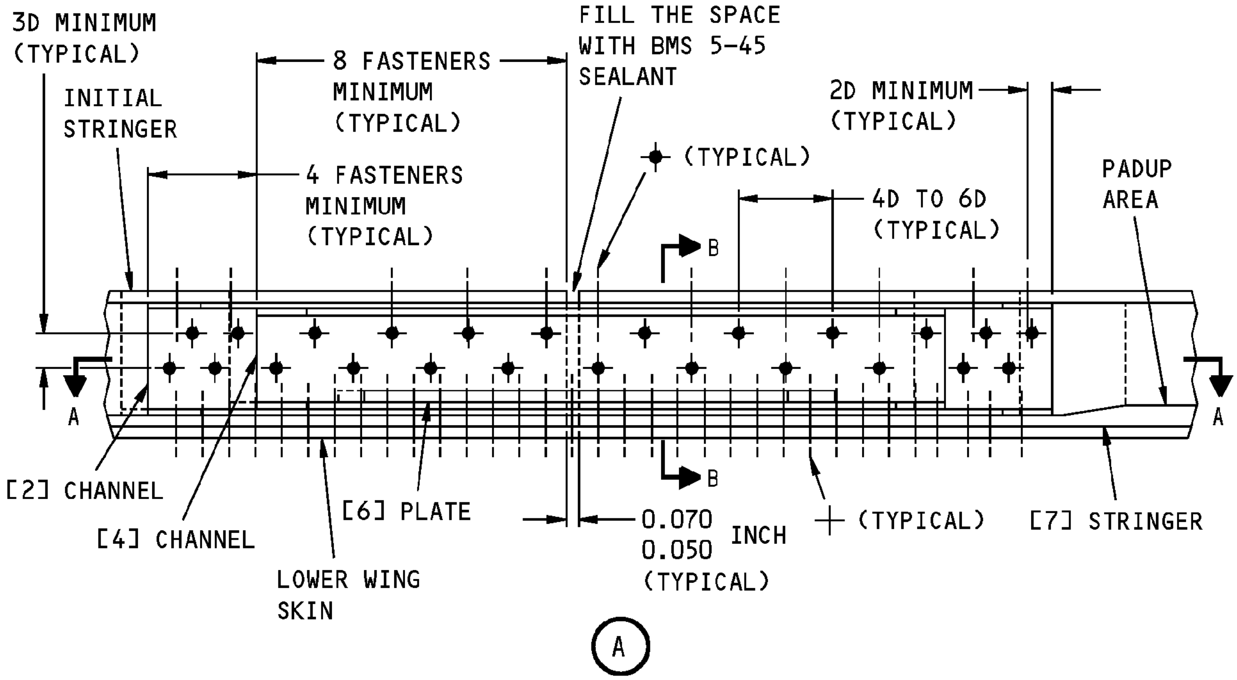
N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

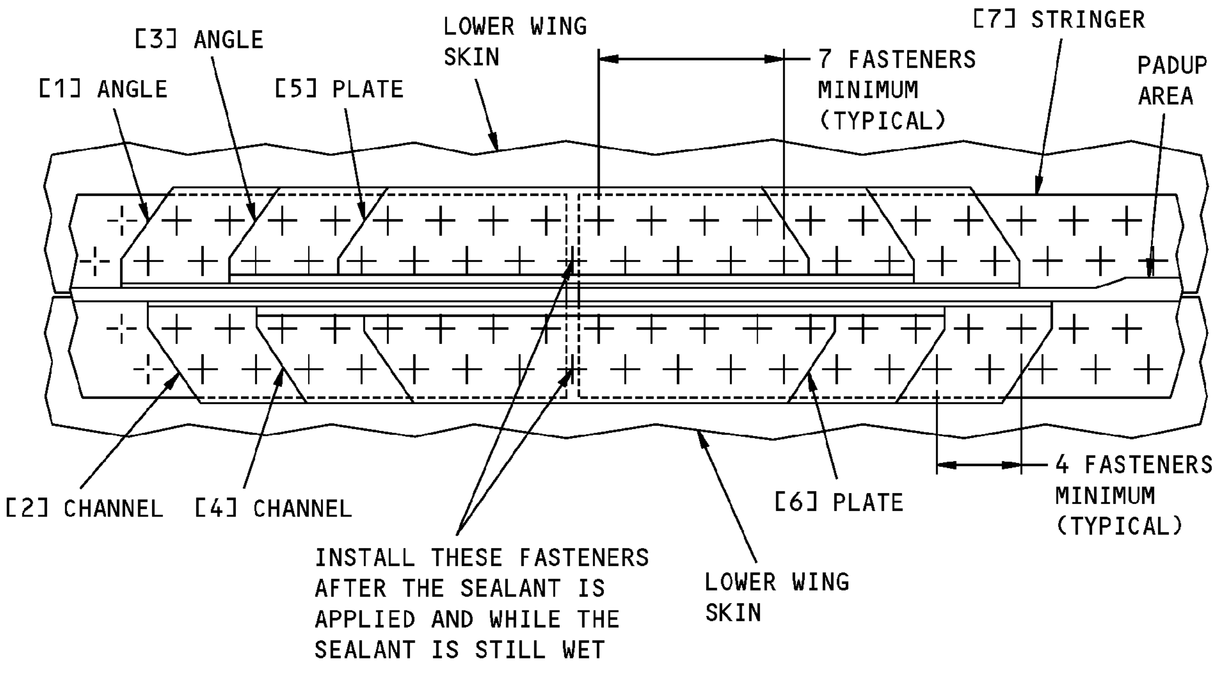
(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.



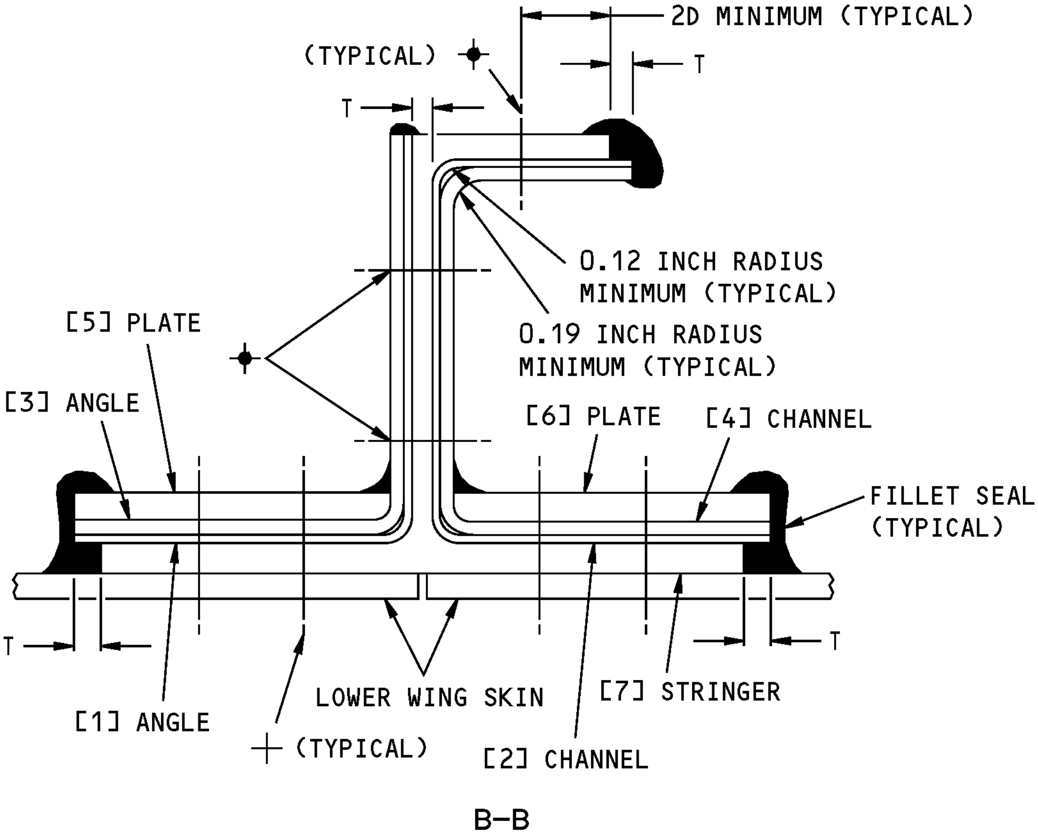

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 3)**




A-A

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 3)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

-●- REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

**Layout of the Repair Parts**
**Figure 202 (Sheet 3 of 3)**

5. **Inspection Instructions**

A. Do a low Frequency Eddy Current (LFEC) inspection at 36,000 flight cycles after you install the repair.

(1) Do the inspection on the external surface of the wing at the stringer repair area.

(a) Inspect at and between each fastener at the repair parts and the repaired area of the stringer.

(2) Refer to 737 NDT Part 6, 51-00-00, Figures 8 and 9 .

(3) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the inspection again at each 18,000 cycles interval. Refer to NDT 6, 51-00-00, Figures 8 and 9 .

## REPAIR 5 - LOWER SPLICE STRINGER ON THE CENTER WING NOT ADJACENT TO A PADUP AREA

1. **Applicability**

   A. Repair 5 is applicable for damage on a lower splice stringer on the center wing that is not adjacent to a padup area. A typical splice stringer is shown in Typical Splice Stringer Section, Figure 201/REPAIR 5.

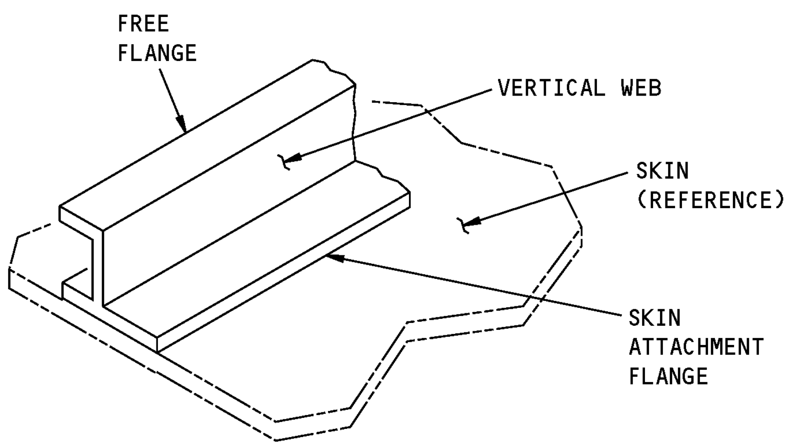
   B. Repair 5 is applicable to locations where there is sufficient clearance between the repair parts and the stringer padup area.

2. **General**

   A. Repair 5 is a Category B repair. Refer to 51-00-06 for the definitions of the different categories of repairs. Refer to the inspection instructions given in Paragraph 5./REPAIR 5

   B. The repair that follows is an alternative to Repair 5:

      (1) Repair 4 is a repair for damage to a lower splice stringer on the center wing that is adjacent to a padup area. Use this repair when there is not sufficient clearance between the repair parts and the stringer padup.




**Typical Splice Stringer Section**
**Figure 201**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |

(Continued)

| Reference | Title |
|---|---|
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-09 | COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 8 | Inspection of Subsurface Cracks in Aluminum Structure |
| 737 NDT Part 6, 51-00-00, Figure 9 | Inspection of Subsurface Cracks at Fastener Holes in Aluminum Structure |

## 4. Repair Instructions

**WARNING**: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION**: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

**CAUTION**: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 5. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 5.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | 1 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [2] | Channel | 1 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [3] | Angle | 1 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [4] | Channel | 1 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [5] | Plate | 1 | Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part |
| [6] | Plate | 1 | Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part |
| [7] | Filler | 1 | Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy. As an alternative, you can machine the part from 2024-T42 aluminum alloy material |

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 5.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.

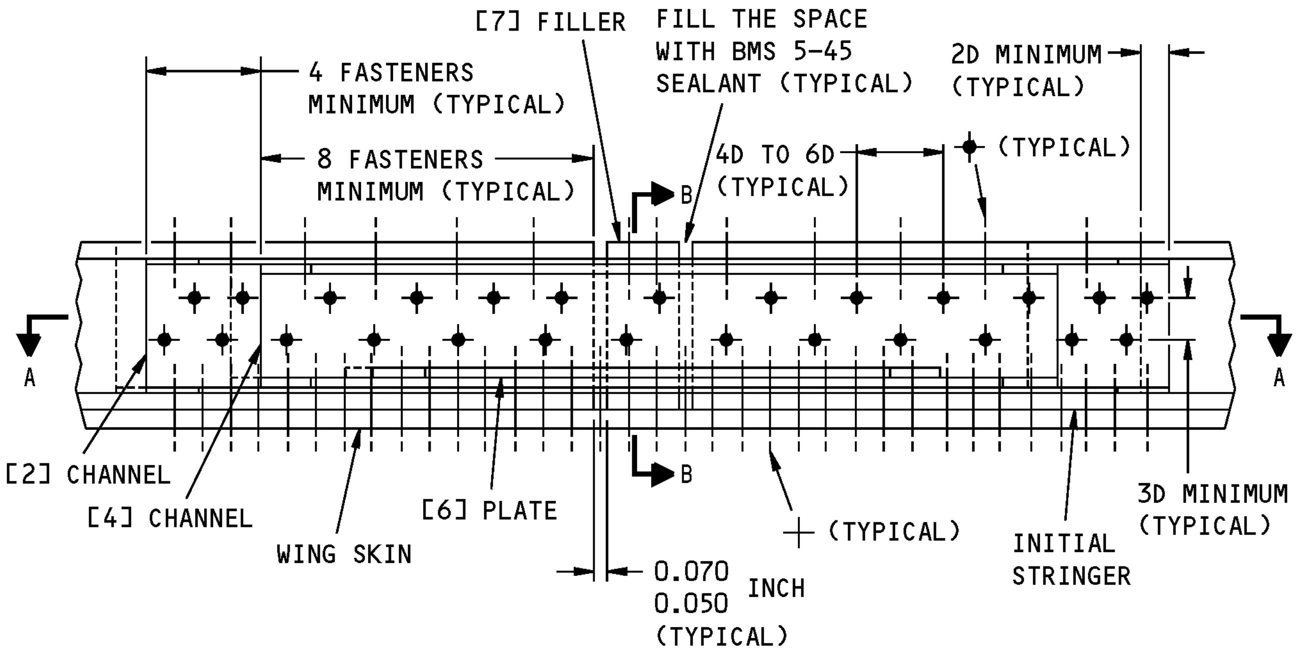
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

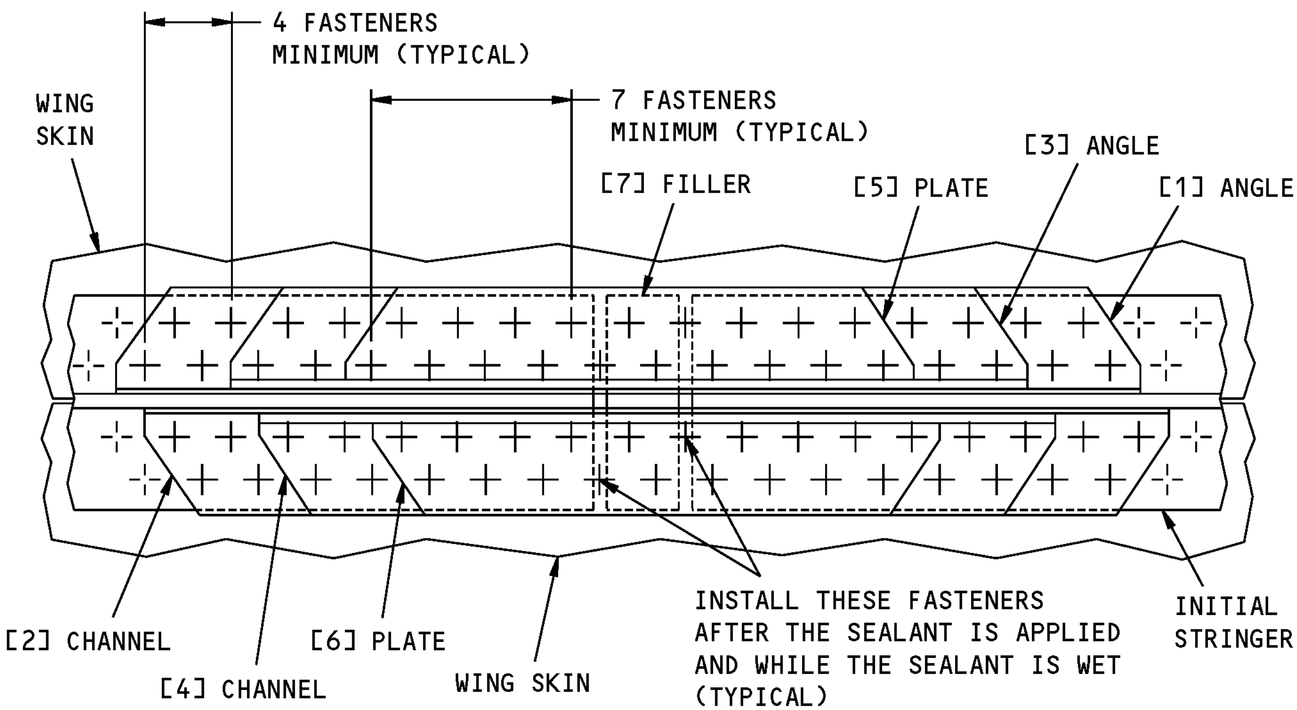
(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

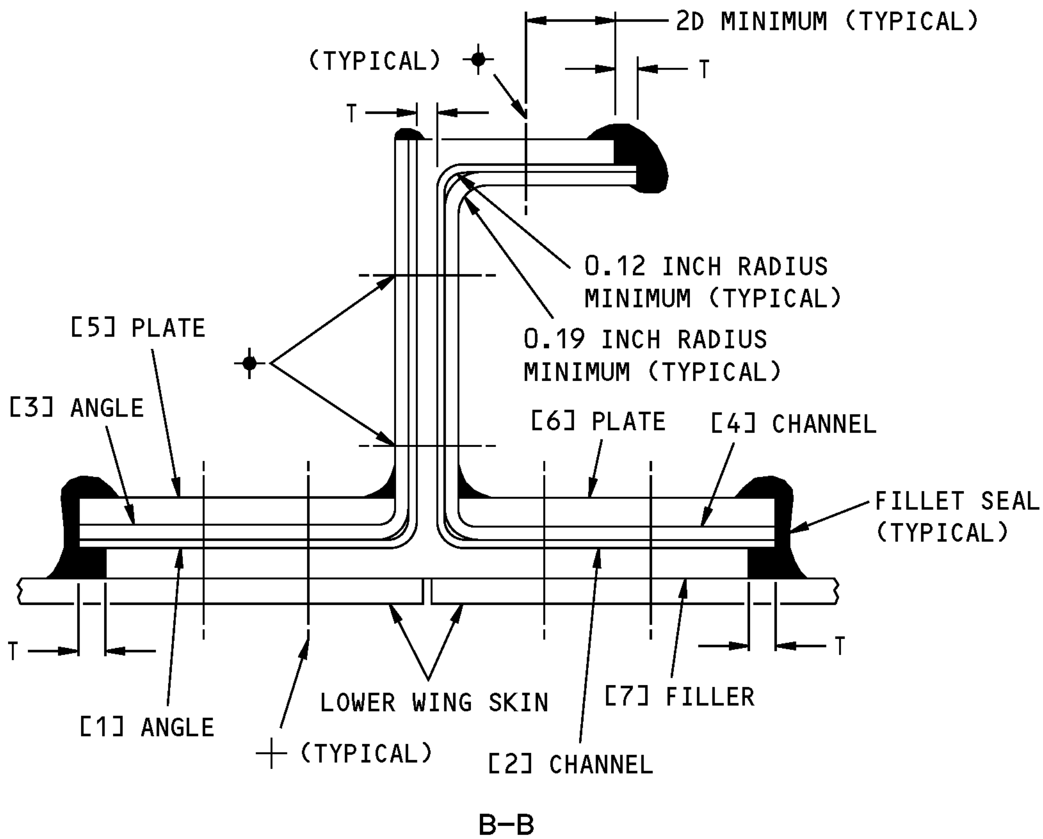
(2) Do a leak test to make sure there are no leaks in the fuel tank.




A-A

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

-◆- REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

## 5. Inspection Instructions

A. Do a Low Frequency Eddy Current (LFEC) inspection at 36,000 flight cycles after you install the repair.

(1) Do the inspection on the external surface of the wing at the stringer repair area.

(a) Inspect at and between each fastener at the repair parts and the repaired area of the stringer.

(2) Refer to 737 NDT Part 6, 51-00-00, Figures 8 and 9 .

(3) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the inspection again at each 18,000 flight cycles interval. Refer to NDT 6, 51-00-00, Figures 8 and 9 .

## REPAIR 6 - LOWER ZEE STRINGER ON THE CENTER WING ADJACENT TO A PADUP AREA

### 1. Applicability

A. Repair 6 is applicable for damage to a lower zee stringer on the center wing that is adjacent to a padup area. A typical zee stringer is shown in Typical Zee Stringer Section, Figure 201/REPAIR 6.

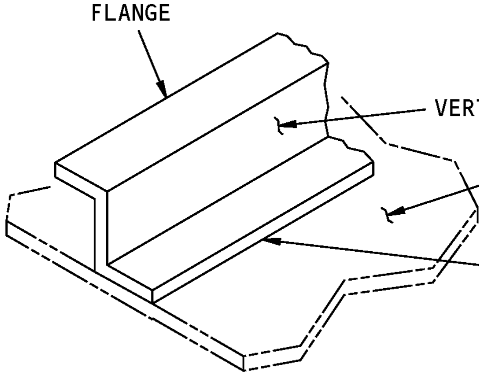
B. Repair 6 is applicable at locations where there is not sufficient clearance between the repair parts and the stringer padup area.

### 2. General

A. Repair 6 is a Category B repair. Refer to 51-00-06 for the definitions of the different categories of repairs. Refer to the inspection instructions given in Paragraph 5./REPAIR 6

B. The repair that follows is an alternative to Repair 6:

(1) Repair 7 is a repair for damage to a lower zee stringer on the center wing that is not adjacent to a padup area. Use this repair when there is sufficient clearance between the repair parts and the stringer padup.




**Typical Zee Stringer Section**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |

(Continued)

| Reference | Title |
|---|---|
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-09 | COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 8 | Inspection of Subsurface Cracks in Aluminum Structure |
| 737 NDT Part 6, 51-00-00, Figure 9 | Inspection of Subsurface Cracks at Fastener Holes in Aluminum Structure |

## 4. Repair Instructions

**WARNING:** MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION:** MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

**CAUTION:** BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 6. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 6.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | 2 | Use 2024-O material, 0.050 inch in thickness. Heat treat to T42 condition after you form the part |
| [2] | Angle | 2 | Use 2024-O material, 0.050 inch in thickness. Heat treat to T42 condition after you form the part |
| [3] | Angle | 2 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [4] | Angle | 2 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [5] | Angle | 2 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [6] | Angle | 2 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [7] | Stringer | 1 | Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy. As an alternative, you can machine the part from 2024-T42 aluminum alloy material |

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] stringer and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 6.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.

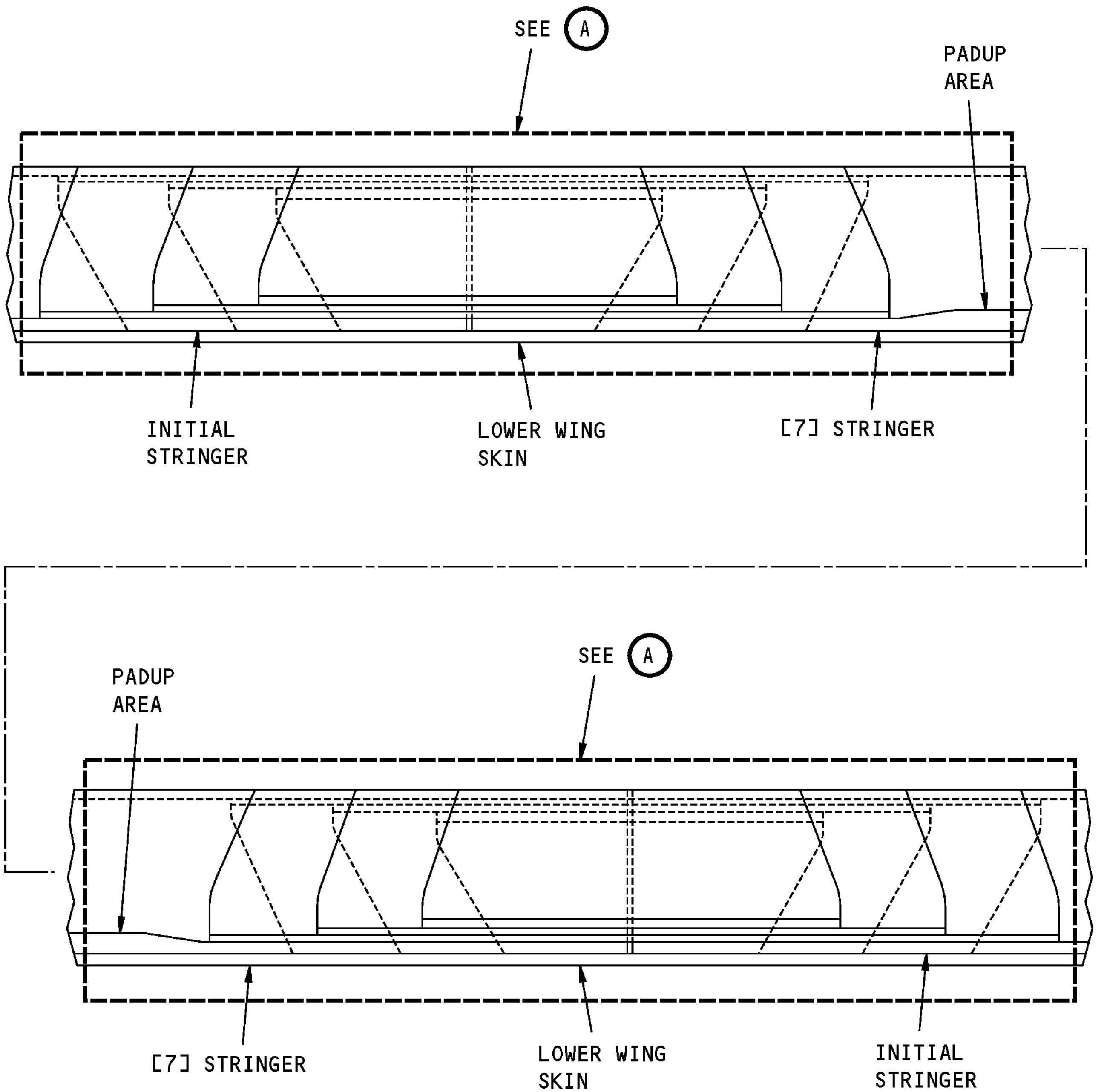
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

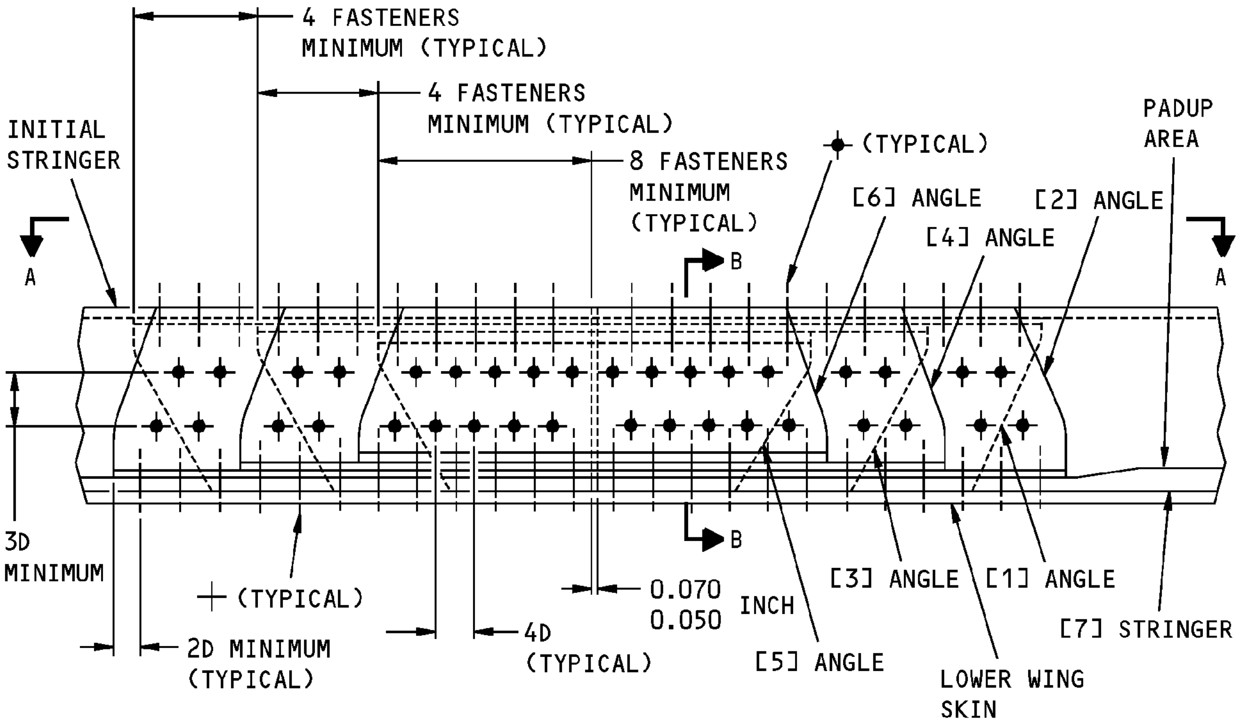
N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

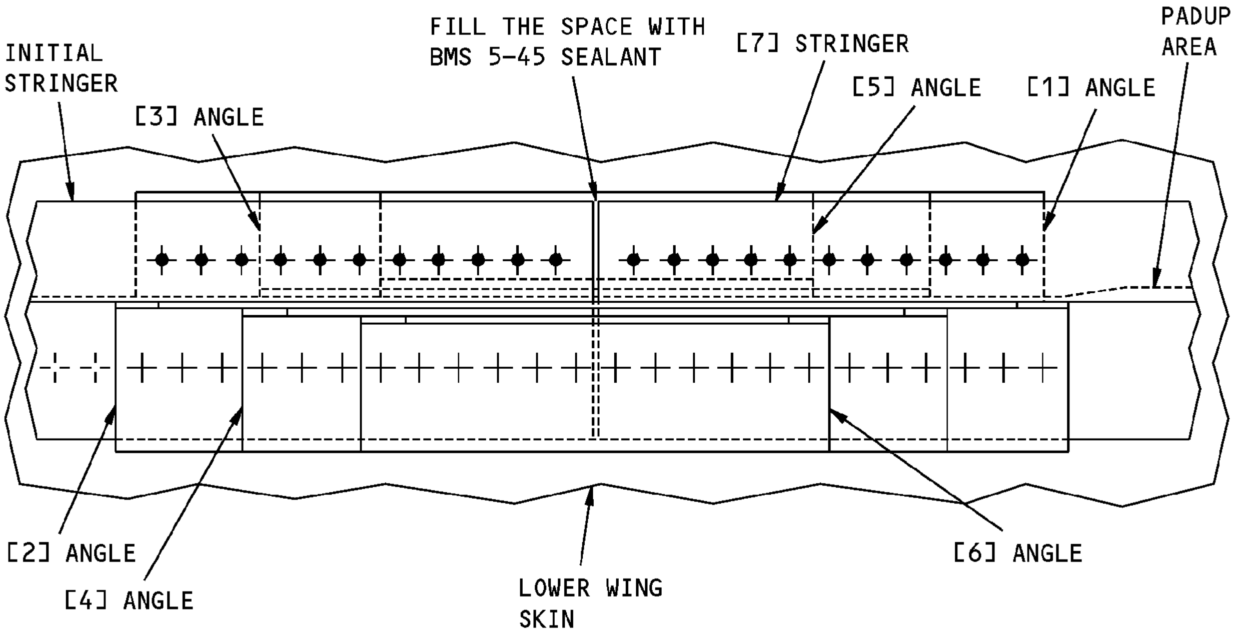
(2) Do a leak test to make sure there are no leaks in the fuel tank.




**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 3)**

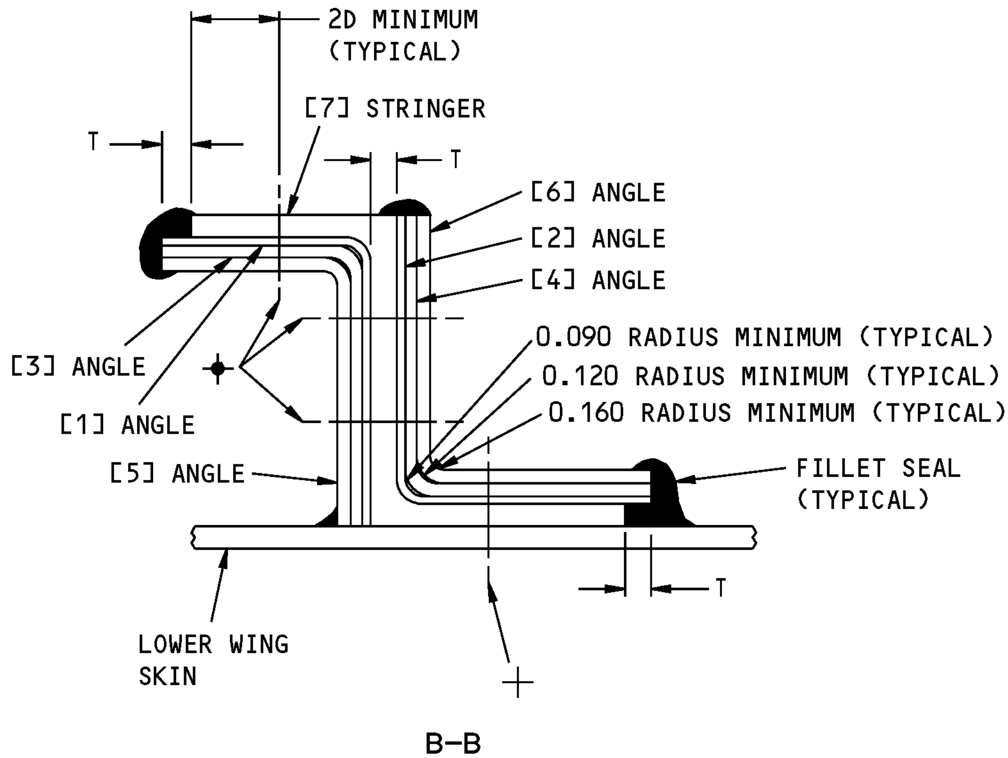



A




A-A

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 3)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

-◆- REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

**Layout of the Repair Parts**
**Figure 202 (Sheet 3 of 3)**

5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection at 36,000 flight cycles after you install the repair.

(1) Inspect at and between each fastener at the repair parts and the repaired area of the stringer.

(2) Refer to 737 NDT Part 6, 51-00-00, Figures 8 and 9 .

(3) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the inspection again at each 18,000 flight cycles interval. Refer to NDT 6, 51-00-00, Figures 8 and 9 .

## REPAIR 7 - LOWER ZEE STRINGER ON THE CENTER WING NOT ADJACENT TO A PADUP AREA

1. **Applicability**

   A. Repair 7 is applicable for damage to a lower zee stringer on the center wing that is not adjacent to a padup area. A typical zee stringer is shown in Typical Zee Stringer Section, Figure 201/REPAIR 7.

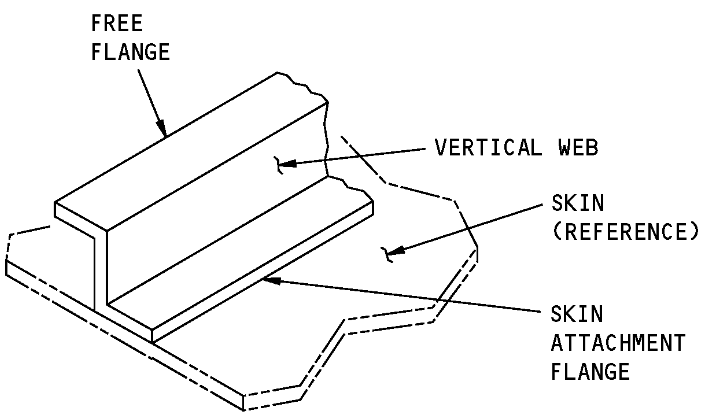
   B. Repair 7 is applicable at locations where there is sufficient clearance between the repair parts and the stringer padup area.

2. **General**

   A. Repair 7 is a Category B repair. Refer to 51-00-06 for the definitions of the different categories of repairs. Refer to the inspection instructions given in Paragraph 5./REPAIR 7

   B. The repair that follows is an alternative to Repair 7:

      (1) Repair 6 is a repair for damage to a lower zee stringer on the center wing that is adjacent to a padup area. Use this repair when there is not sufficient clearance between the repair parts and the stringer padup.




**Typical Zee Stringer Section
Figure 201**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |

(Continued)

| Reference | Title |
|---|---|
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-09 | COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 8 | Inspection of Subsurface Cracks in Aluminum Structure |
| 737 NDT Part 6, 51-00-00, Figure 9 | Inspection of Subsurface Cracks at Fastener Holes in Aluminum Structure |

## 4. Repair Instructions

**WARNING:** MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION:** MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

**CAUTION:** BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 7. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 7.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Angle | 1 | Use 2024-O material, 0.050 inch in thickness. Heat treat to T42 condition after you form the part |
| [2] | Angle | 1 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [3] | Angle | 1 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [4] | Angle | 1 | Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part |
| [5] | Angle | 1 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [6] | Angle | 1 | Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part |
| [7] | Filler | 1 | Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy |

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 7.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.

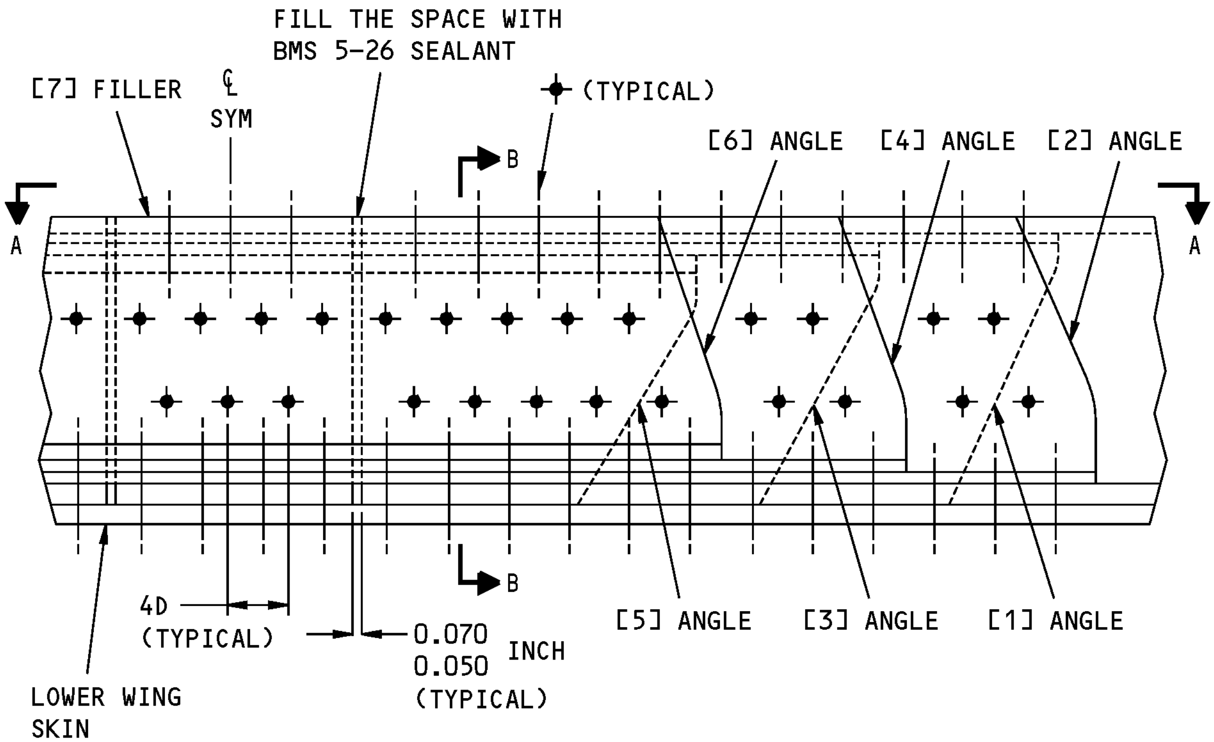
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

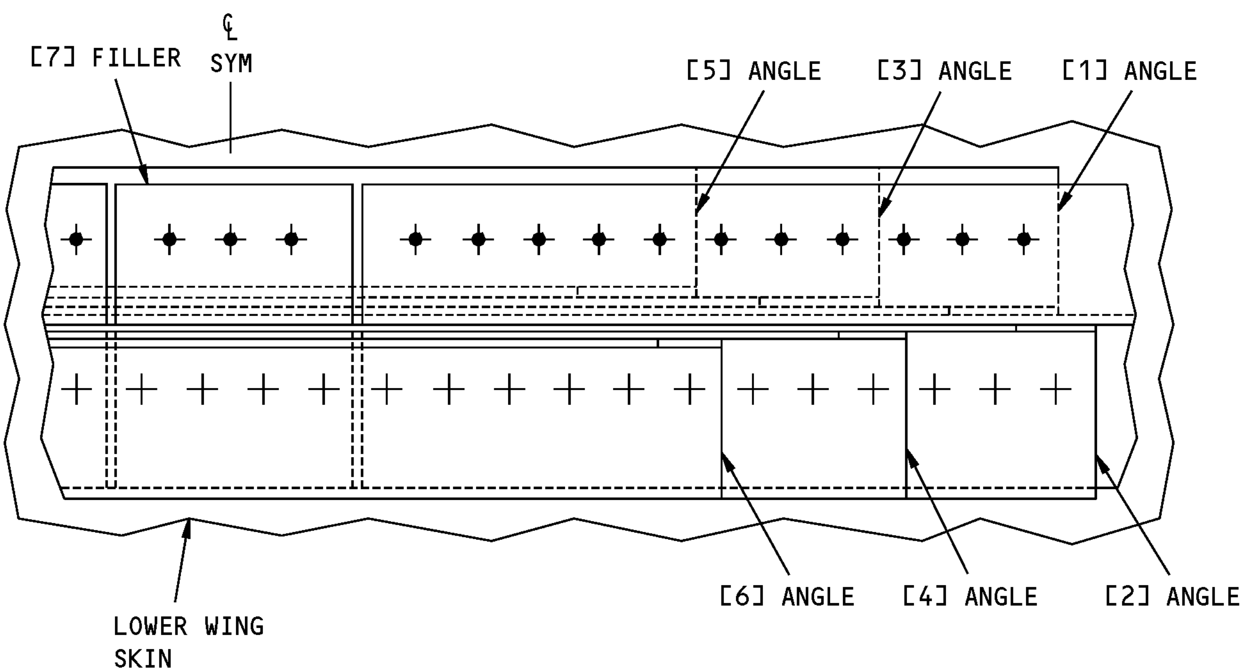
(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

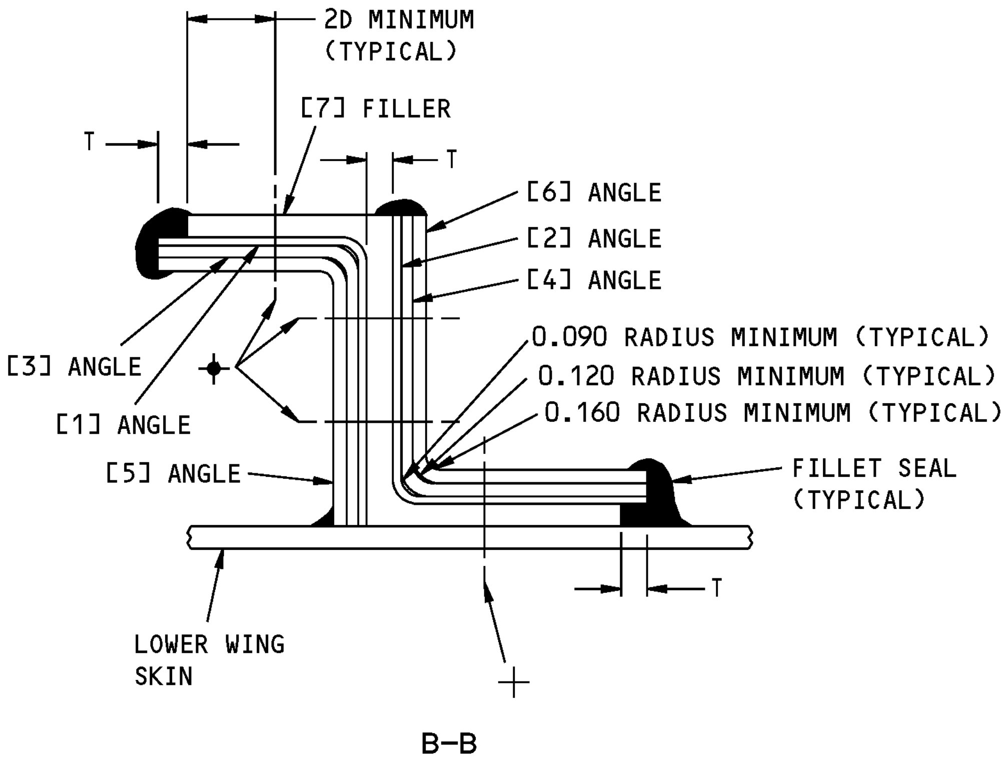
(2) Do a leak test to make sure there are no leaks in the fuel tank.




A-A

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




FASTENER SYMBOLS

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

⌖ REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection at 36,000 flight cycles after you install the repair.

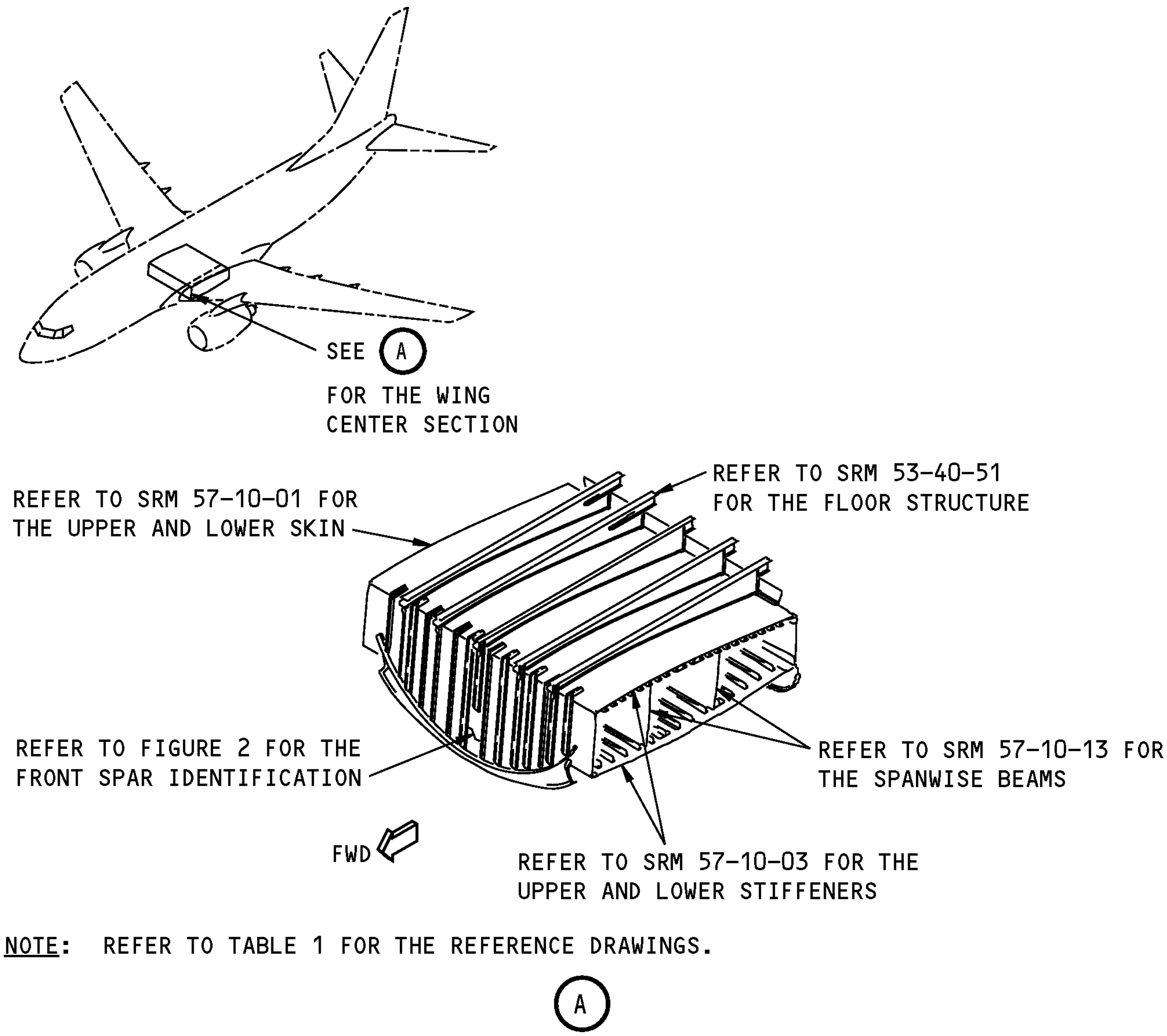
(1) Inspect at and between each fastener at the repair parts and the repaired area of the stringer.

(2) Refer to 737 NDT Part 6, 51-00-00, Figures 8 and 9 .

(3) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the inspection again at each 18,000 flight cycles interval. Refer to NDT 6, 51-00-00, Figures 8 and 9 .

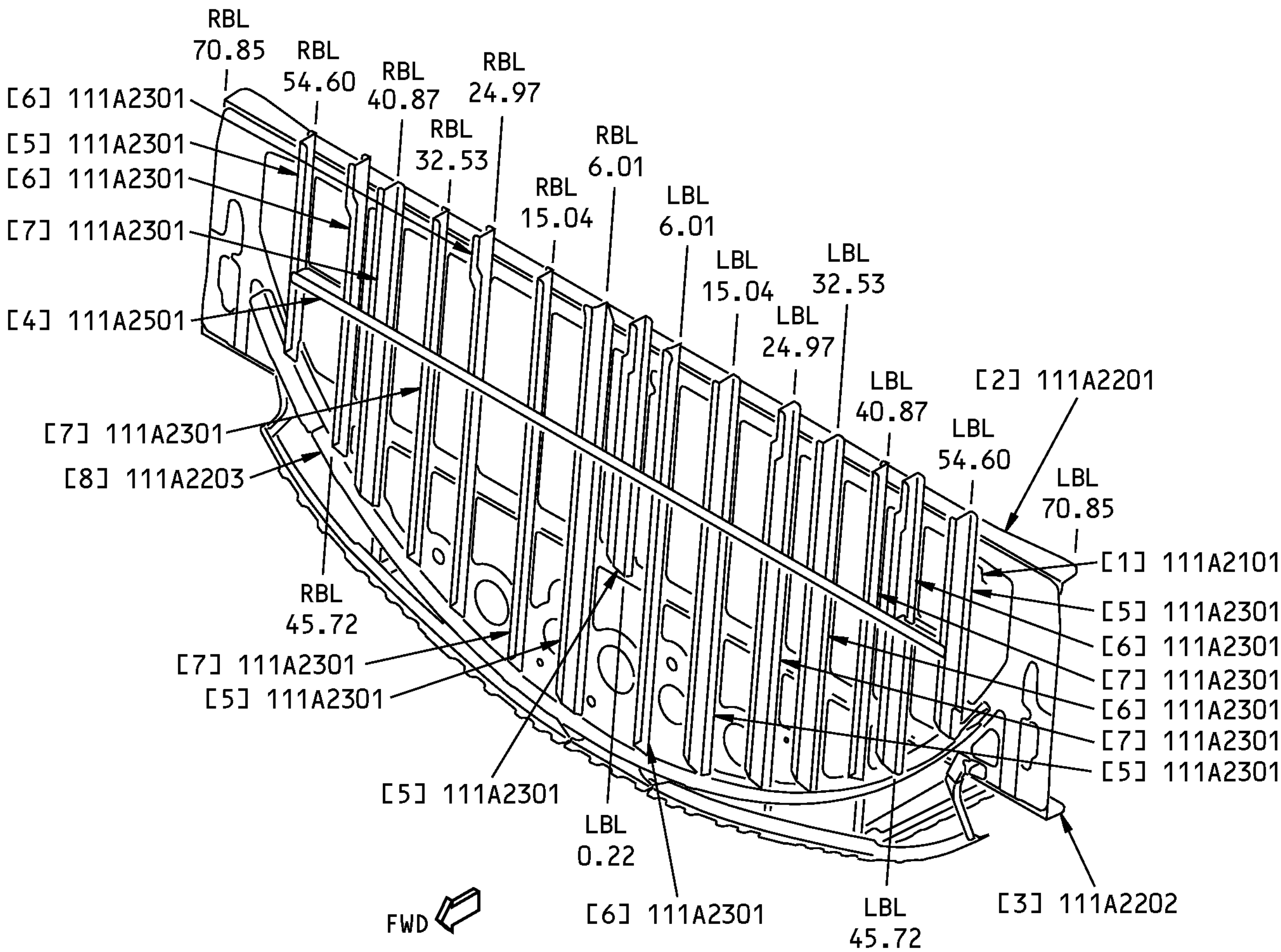
## IDENTIFICATION 1 - WING CENTER SECTION FRONT SPAR




**Wing Center Section Front Spar Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 111A2000 | Front Spar Installation - Wing Center Section |
| 111A2001 | Front Spar Assembly - Wing Center Section |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

VIEW LOOKING AFT

A

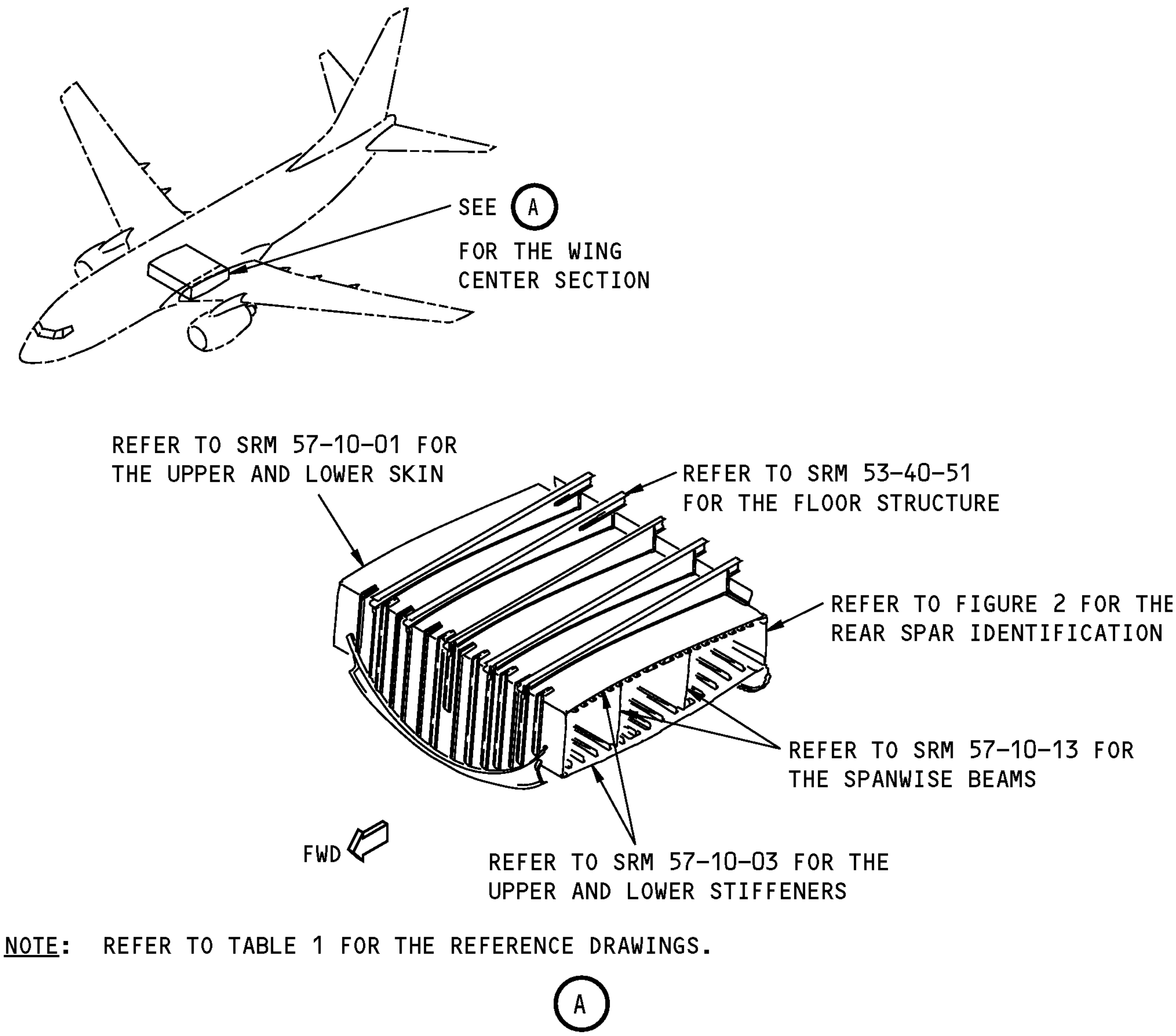
**Wing Center Section Front Spar Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Web | 0.250 (6.35) | 2024-T351 plate as given in QQ-A-250/4 | |
| [2] | Upper Chord | | BAC1514-3221 7150-T77511 extrusion as given in BMS 7-306 | |
| [3] | Lower Chord | | BAC1514-3222 2224-T3511 extrusion as given in BMS 7-255 | |
| [4] | Stabilizer Strap | 0.040 (1.02) | 7075-T6 clad sheet as given in QQ-A-250/13 | |
| [5] | Stiffener | | BAC1506-3045 7150-T77511 extrusion as given in BMS 7-306 | |
| [6] | Stiffener | | BAC1517-2794 7150-T77511 extrusion as given in BMS 7-306 | |
| [7] | Stiffener | | BAC1517-2793 7150-T77511 extrusion as given in BMS 7-306 | |
| [8] | Bulkhead Chord | | BAC1514-3131 2024-T42 extrusion as given in QQ-A-200/3 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

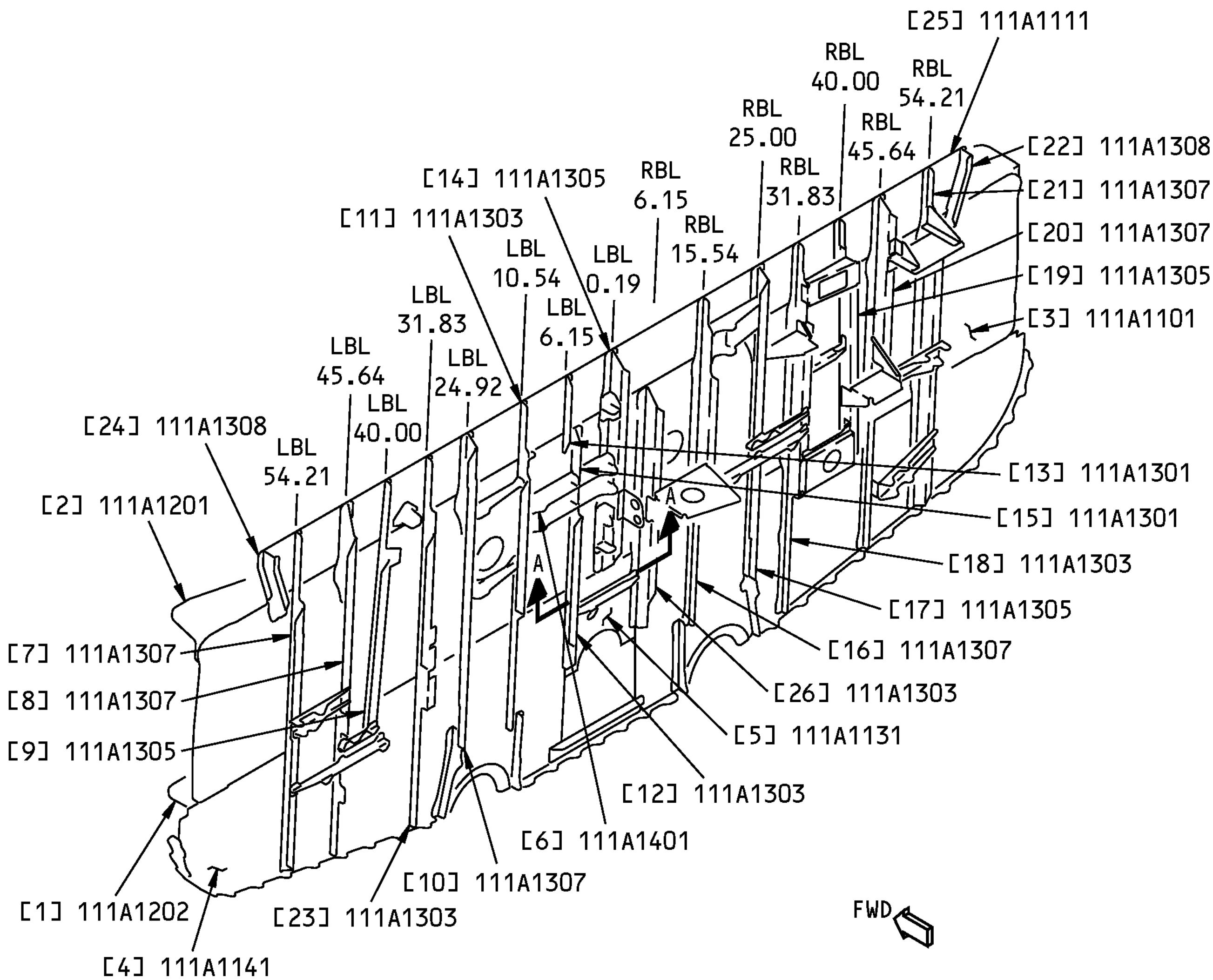
## IDENTIFICATION 2 - WING CENTER SECTION REAR SPAR




**Wing Center Section Rear Spar Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 111A1000 | Rear Spar Installation - Wing Center Section |
| 111A1001 | Rear Spar Assembly - Wing Center Section |



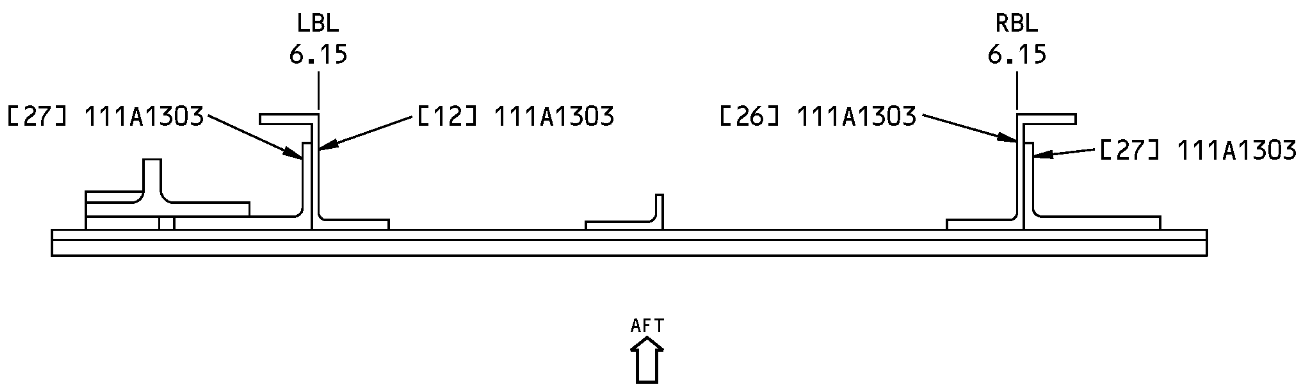

NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

REAR VIEW

CENTER SECTION REAR SPAR

A

**Wing Center Section Rear Spar Identification**
**Figure 2 (Sheet 1 of 2)**




FOR AIRPLANES THAT HAVE BEEN MODIFIED
BY SB 737-57-1253

SECTION A-A

**Wing Center Section Rear Spar Identification**
**Figure 2 (Sheet 2 of 2)**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Lower Chord | | BAC1514-3220 2224-T3511 extrusion as given in BMS 7-255 | |
| [2] | Upper Chord | | BAC1514-3331 7150-T77511 extrusion as given in BMS 7-306 | |
| [3] | Web | | 2324-T39 plate as given in BMS 7-254 | |
| [4] | Lower Left and Right Webs | | Glass Fiber Reinforced Plastic (GFRP) laminate as given in BMS 8-139, Type 181, 4 plies (Optional: BMS 8-79, Class I, Type 181, 4 plies) | |
| [5] | Lower Center Web | 0.063 (1.6) | 2024-T3 clad sheet as given in QQ-A-250/5 | |
| [6] | Support Fitting | | 7050-T7451 plate as given in BMS 7-323 and AMS 4050 | |
| [7] | Stiffener | | BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306 | |
| [8] | Stiffener | | BAC1506-4364 7150-T77511 extrusion as given in BMS 7-306 | |
| [9] | Stiffener | | BAC1506-4361 7150-T77511 extrusion as given in BMS 7-306 | |
| [10] | Stiffener | | BAC1506-4363 7150-T77511 extrusion as given in BMS 7-306 | |
| [11] | Stiffener | | BAC1517-2776 7150-T77511 extrusion as given in BMS 7-306. | |
| [12] | Stiffener LBL 6.15 | | BAC1517-2776 7150-T77511 extrusion as given in BMS 7-306 or BAC1517-2780 altered to match cross section of BAC1517–2276. As an optional BAC1518–1152 7055–T77511 extrusion as given in BMS 7-308 | Airplane line numbers 1 thru 439 |
| [12] | Stiffener LBL 6.15 | | 2024-T351 plate as given in QQ-A-250/4 or 2024-T351 extruded bar as given in QQ-A-200/3, machined to BAC1517-2776 shape. | Airplane line numbers 440 and on |
| [13] | Stiffener | | BAC1503-4223 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [14] | Stiffener | | BAC1506-4361 7150-T77511 extrusion as given in BMS 7-306 | |
| [15] | Stiffener | | BAC1514-1160 7150-T77511 extrusion, altered, as given in BMS 7-306 | |
| [16] | Stiffener | | BAC1517-2779 7150-T77511 extrusion as given in BMS 7-306 | |
| [17] | Stiffener | | BAC1517-4362 7150-T77511 extrusion as given in BMS 7-306 | |
| [18] | Stiffener | | BAC1517-2778 7150-T77511 extrusion as given in BMS 7-306 | |
| [19] | Stiffener | | BAC1506-436 7150-T77511 extrusion as given in BMS 7-306 | |
| [20] | Stiffener | | BAC1506-4364 7150-T77511 extrusion as given in BMS 7-306 | |

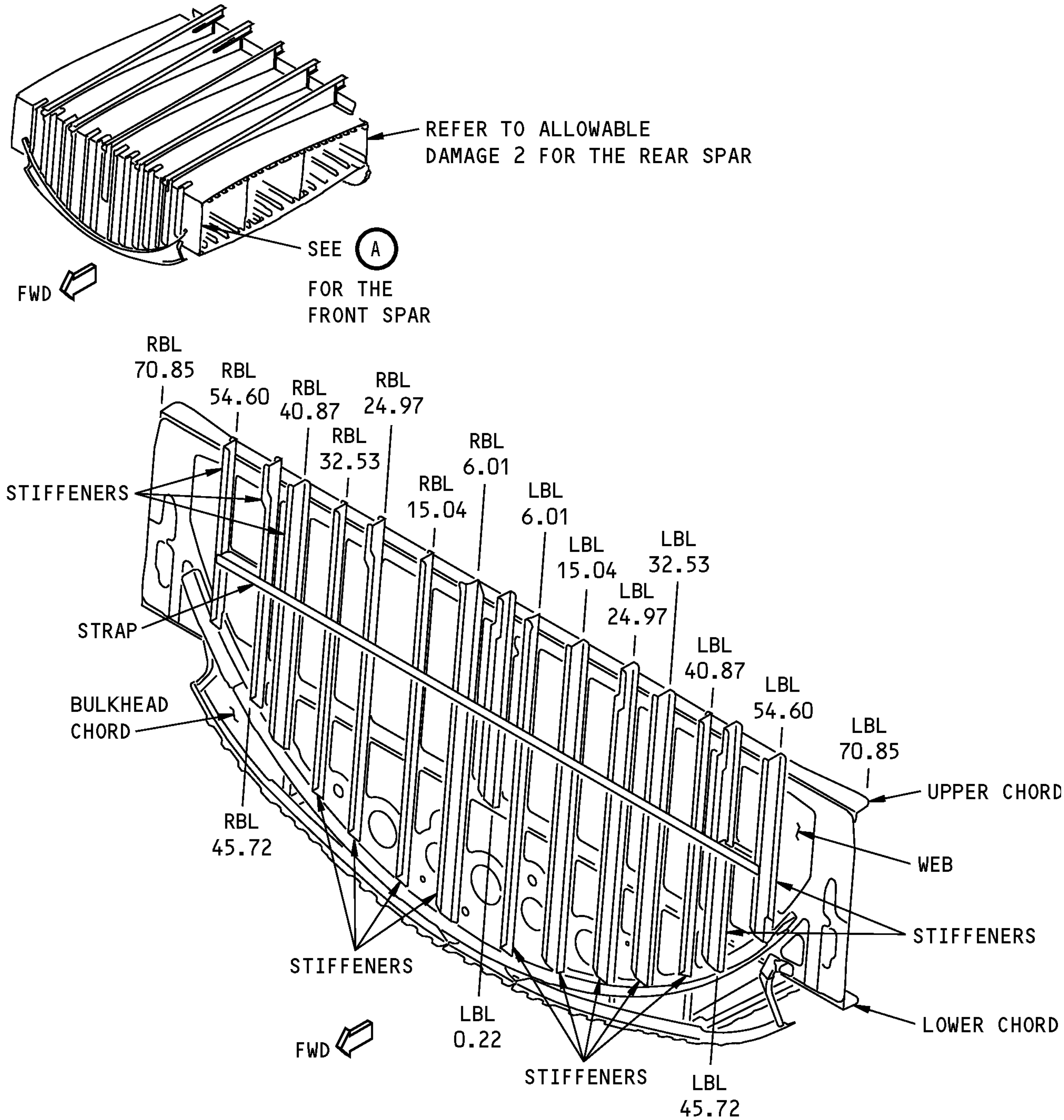
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [21] | Stiffener | | BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306 | |
| [22] | Stiffener | | 7050-T7451 plate as given in AMS 4050 | |
| [23] | Stiffener | | BAC1517-2778 7150-T77511 extrusion as given in BMS 7-306 | |
| [24] | Stiffener | | 7050-T7451 plate as given in AMS 4050 | |
| [25] | Vertical Pressure Web | | 2024-T3 bare sheet as given in QQ-A-250/4 | |
| [26] | Stiffener, RBL 6.14 | | BAC1517-2778 7150-T77511 extrusion as given in BMS 7-306 or BAC1517-2779 altered to match cross section of BAC1517-2778. As an optional BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306 or BAC1518-1152 7055-T77511 extrusion as given in BMS 7-308. | Airplane line numbers 1 thru 439 |
| [26] | Stiffener, RBL 6.14 | | 2024-T351 plate as given in QQ-A-250/4 or 2024-T3511 extruded bar as given in QQ-A-200/3, machined to BAC1517-2776 shape. | Airplane line numbers440 and on |
| [27] | Angle (2) | | 7050-T7451 plate as given in BMS 7-323 Type III, machined to make an angle. | Airplane line numbers 1 thru 321 that have been modified by SB 57–1253 |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## ALLOWABLE DAMAGE 1 - WING CENTER SECTION FRONT SPAR

### 1. Applicability

A. This subject gives the allowable damage limits for the wing center section front spar as shown in Wing Center Section Front Spar Location, Figure 101/ALLOWABLE DAMAGE 1.




NOTE: ALL PARTS ARE MADE OF ALUMINUM.

VIEW LOOKING AFT
A

**Wing Center Section Front Spar Location**
**Figure 101**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the spar chords or the web.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(3) Apply a layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-41-02.

3. **References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

4. **Allowable Damage Limits**

A. Chords (Except the Bulkhead Chord)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and F .

(3) Dents are not permitted.

(4) Holes and punctures are not permitted.

B. Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and G .

(b) You can remove up to 20 percent of the total cross sectional area of the web.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , E , F , and G .

(b) You can remove up to 20 percent of the total cross sectional area of the web.

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail H .

(4) Holes and Punctures:

(a) Damage is permitted if:

1) The damage is a maximum of 0.25 inch diameter after the damage is removed.

2) The damage is in an area away from a padup.

3) The damage is a minimum of 1.00 inch away from a hole, an edge, or other damage.

4) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet.

a) Install the rivet wet with BMS 5-26 sealant.

C. Stiffeners

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and F .

(3) Dents are not permitted.

(4) Holes and Punctures:

(a) The damage is permitted if:

1) The damage is a maximum of 0.25 inch diameter after the damage is removed.

2) The damage is a minimum of 0.30 inch from the edge of the part.

3) The damage is a minimum of 0.75 inch from a hole or other damage.

4) The damage is to a maximum of 4 holes for each stiffener.

5) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet.

a) Install the rivet wet with BMS 5-26 sealant.

D. Strap

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and I .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove edge damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , E , F , and I .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail H .

(4) Holes and punctures are not permitted.

E. Bulkhead Chord

(1) Cracks:

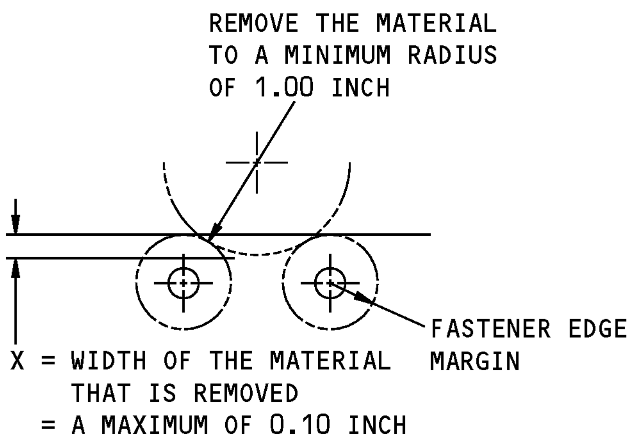
(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and I .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , E , F , and I .

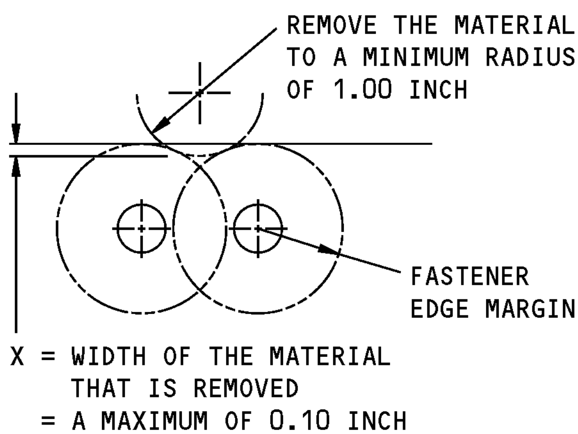
(3) Dents are not permitted.

(4) Holes and punctures are not permitted.




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

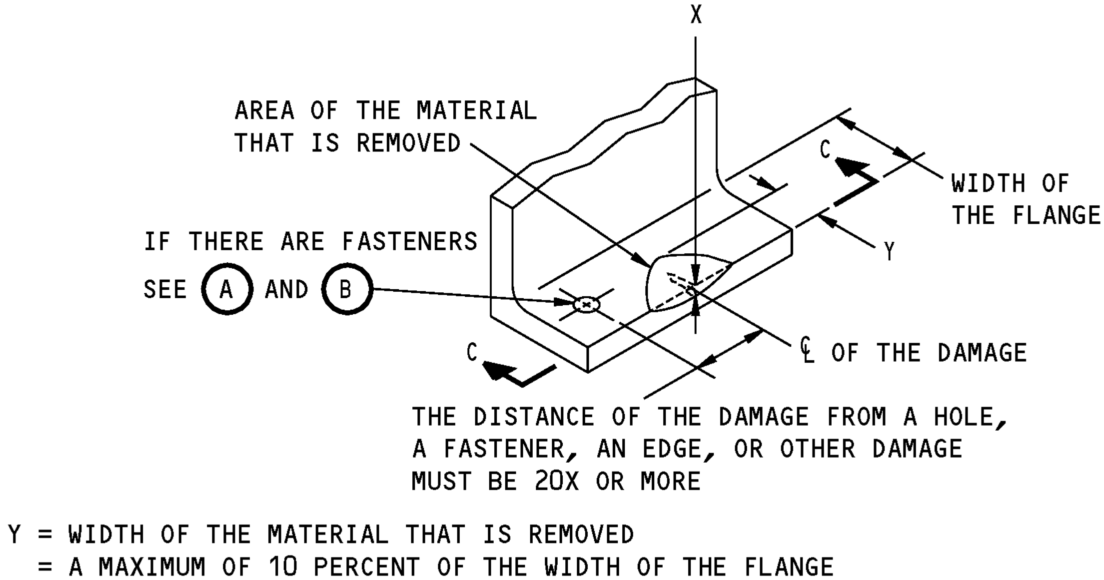
A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

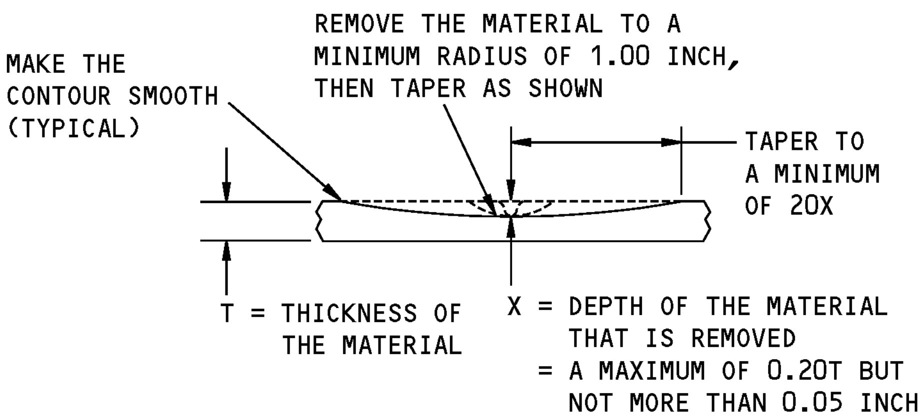
B

**Allowable Damage Limits**
**Figure 102 (Sheet 1 of 5)**



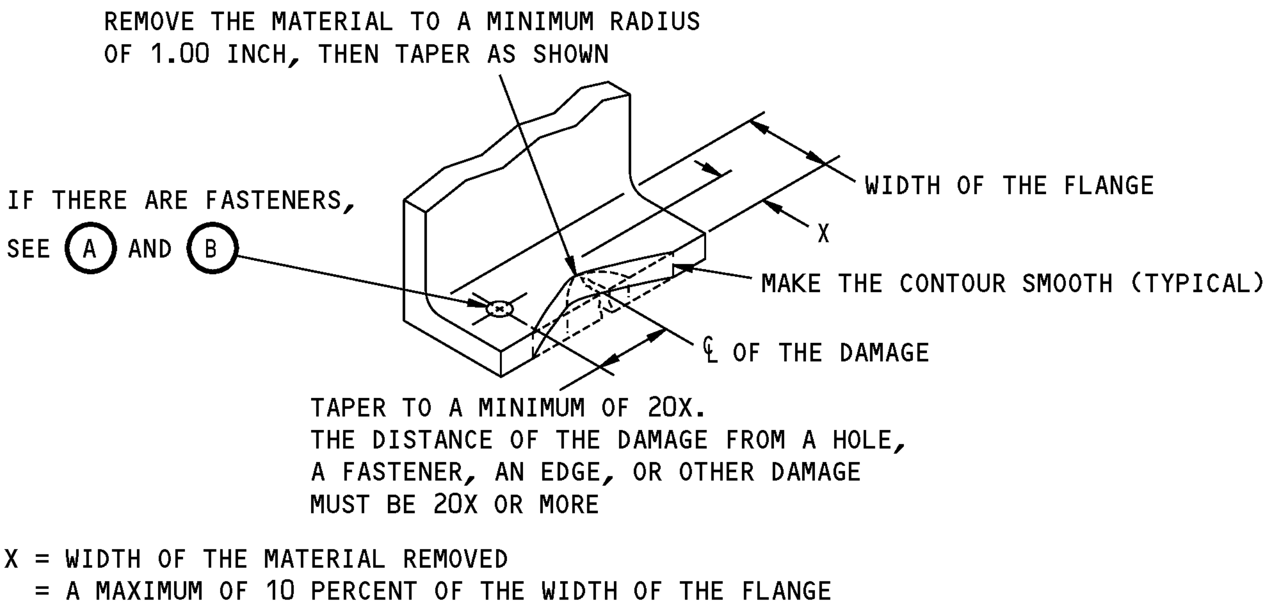

REMOVAL OF DAMAGED MATERIAL ON A
SURFACE AT AN EDGE OF AN EXTRUDED PART

C



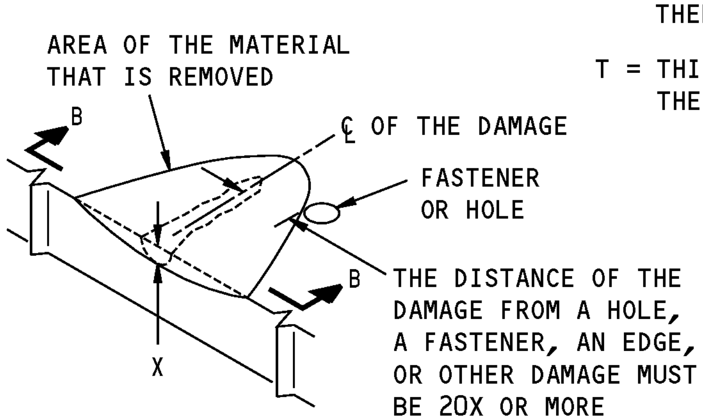

A-A

**Allowable Damage Limits**
**Figure 102 (Sheet 2 of 5)**



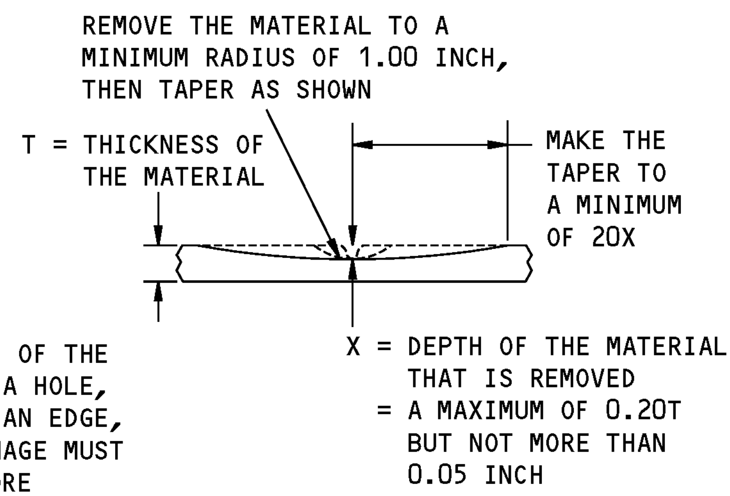

REMOVAL OF DAMAGED MATERIAL
AT AN EDGE OF AN EXTRUDED PART

D



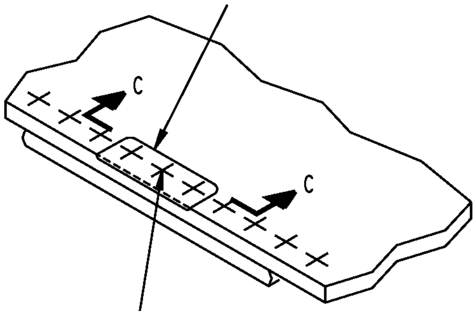

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

E



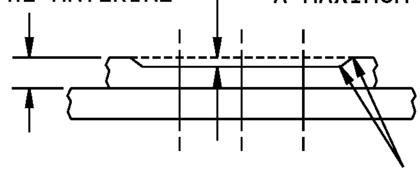

B-B

**Allowable Damage Limits**
**Figure 102 (Sheet 3 of 5)**

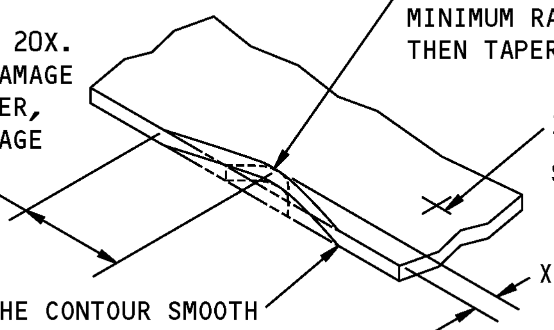



REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

F




C-C

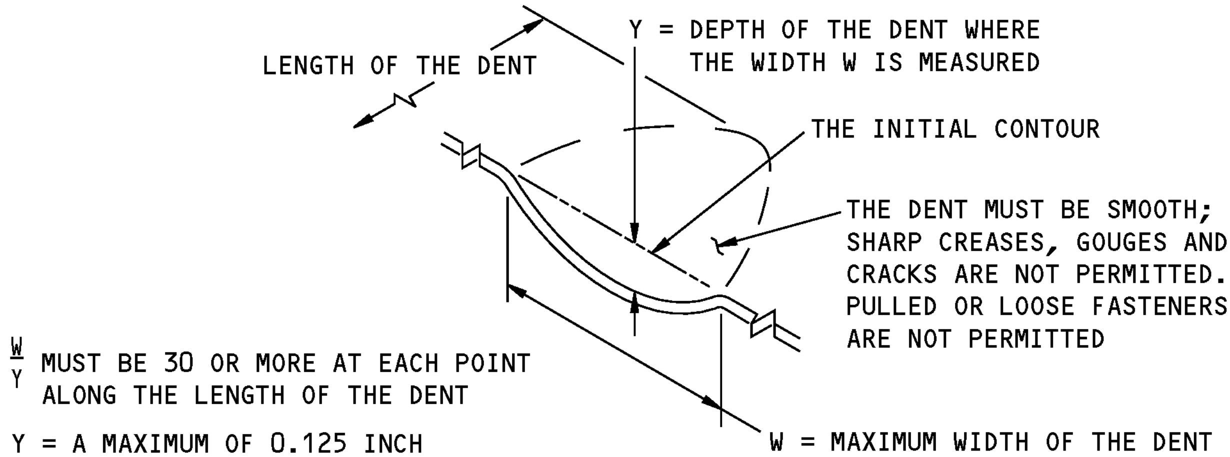



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A WEB

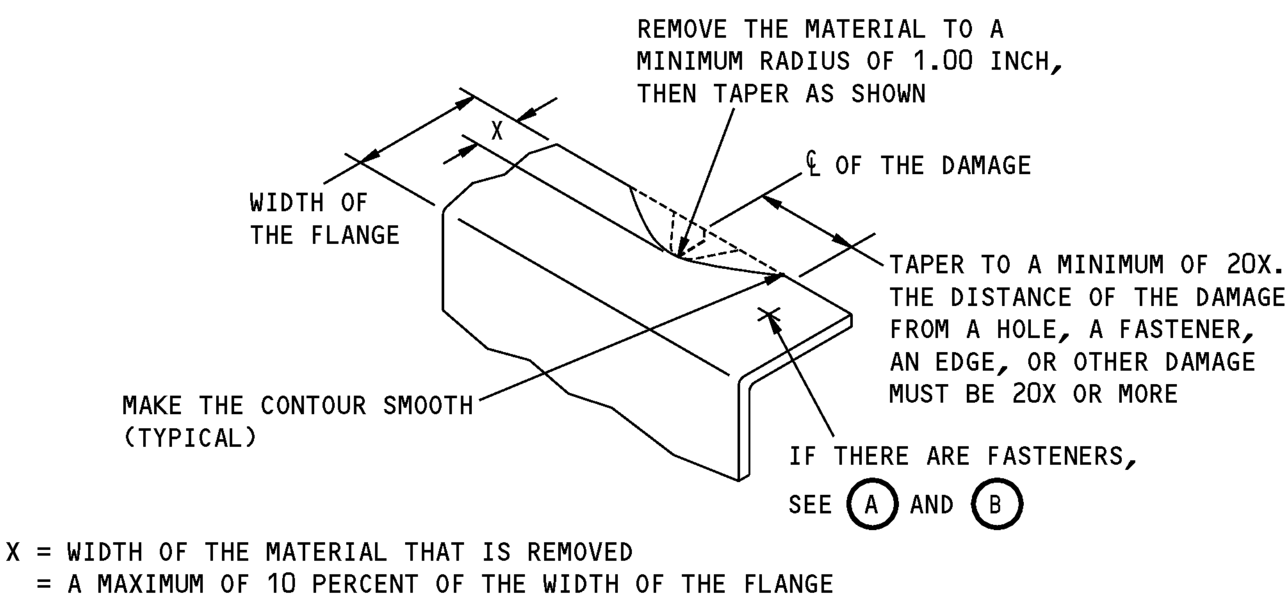
G

**Allowable Damage Limits**
**Figure 102 (Sheet 4 of 5)**




DENT THAT IS PERMITTED

H




X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE OF A FORMED PART

I

**Allowable Damage Limits**
**Figure 102 (Sheet 5 of 5)**

## ALLOWABLE DAMAGE 2 - WING CENTER SECTION REAR SPAR

### 1. Applicability

A. Allowable Damage 2 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane. Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

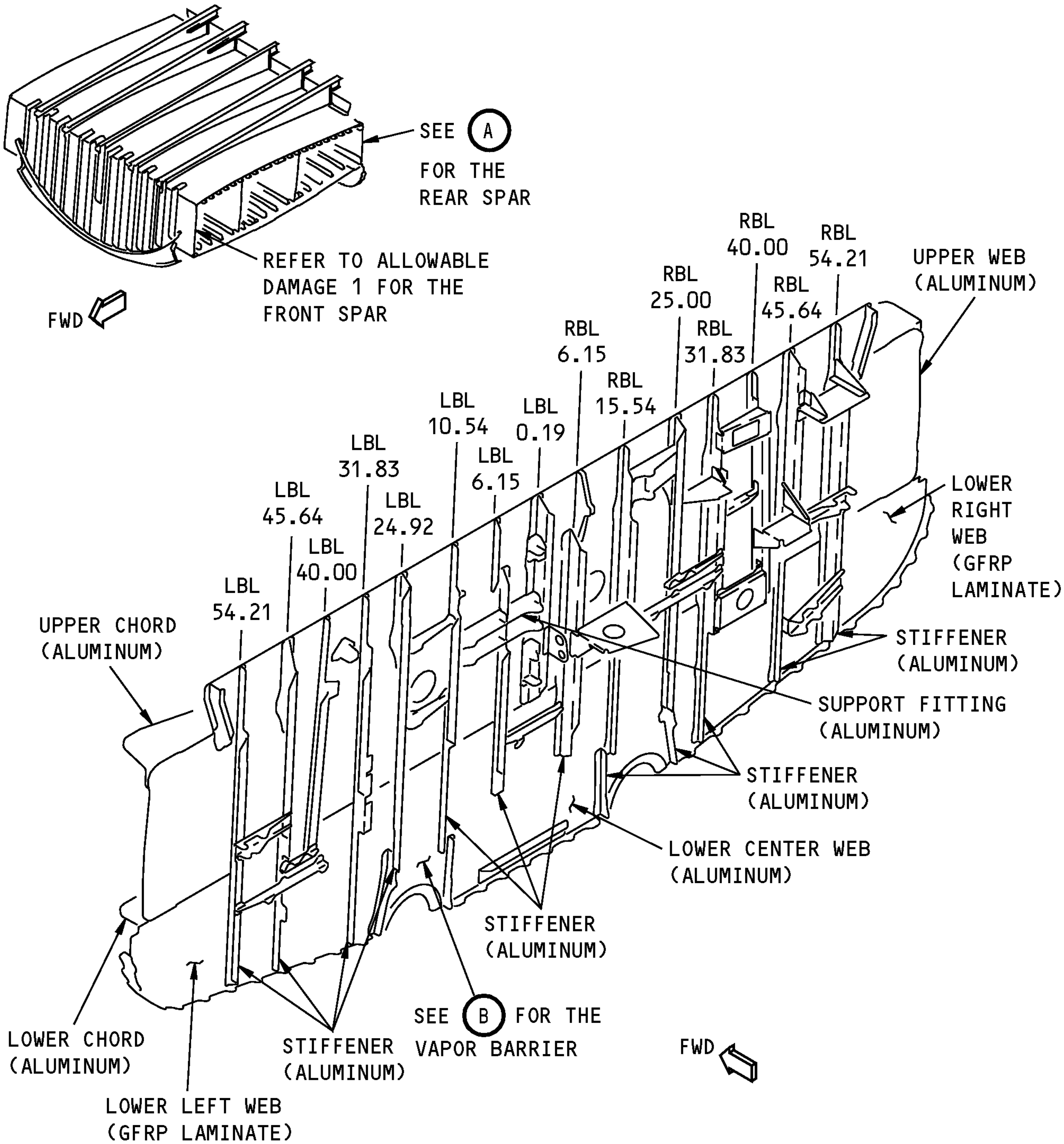
B. Allowable Damage 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

C. This subject gives the allowable damage limits for the wing center section rear spar as shown in Figure 101/ALLOWABLE DAMAGE 2.

D. Allowable Damage 2 is applicable to:

(1) Single cracks in the Vapor Barrier no more than 16 inches long.

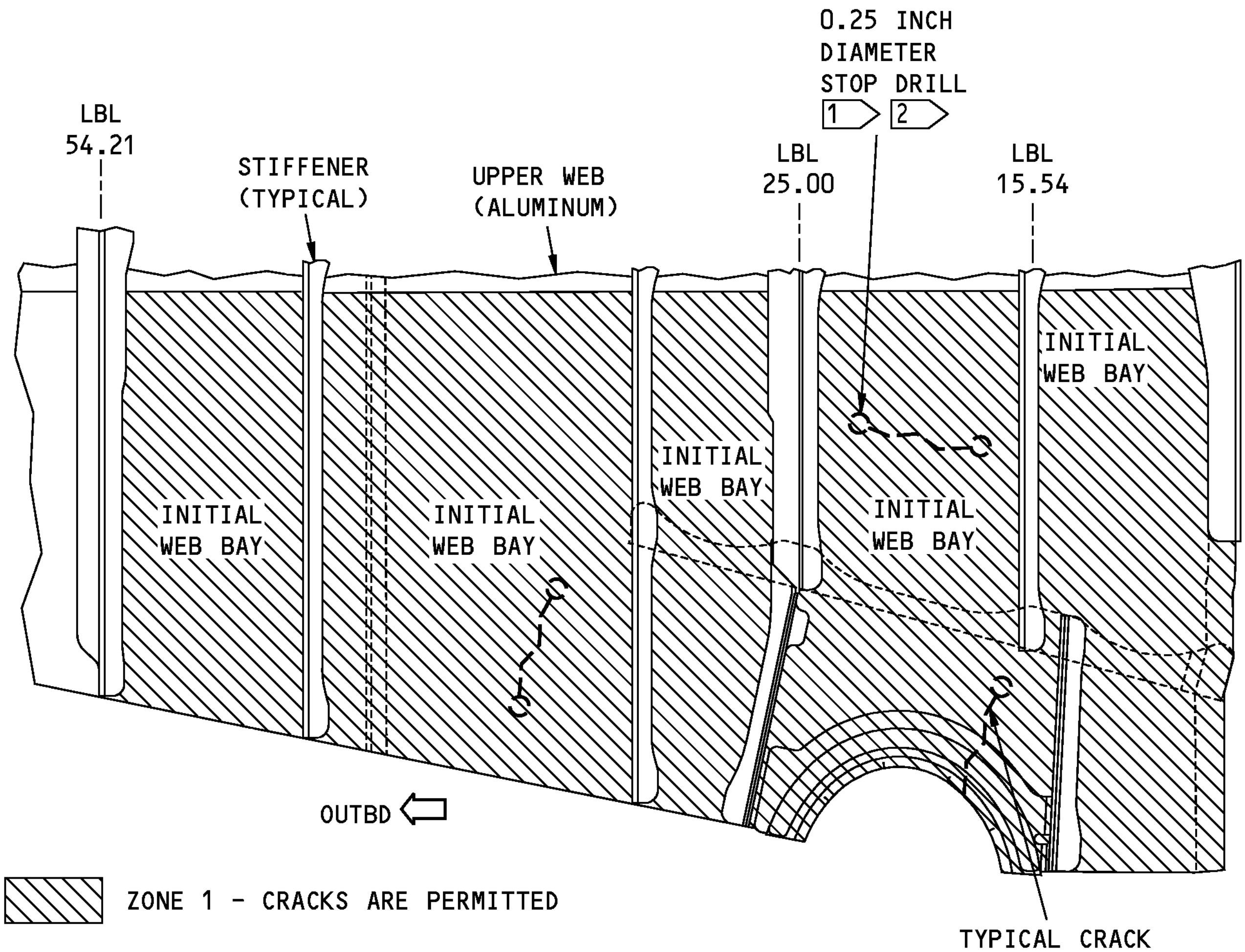
(2) More than one crack in the Vapor Barrier on the left or right of BL 0. The sum of the cracks on either side of BL 0 must not be more than 20 inches long .




NOTE: REFER TO PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

A

**Wing Center Section Rear Spar Allowable Damage**
**Figure 101 (Sheet 1 of 2)**




NOTES

- REFER TO PARAGRAPH 2 OR 4, AS APPLICABLE FOR THE ALLOWABLE DAMAGE LIMITS.

1 MORE THAN ONE CRACK IS PERMITTED ON THE LEFT OR RIGHT OF BL 0.0. THE SUM OF THE LENGTHS OF THE CRACKS ON EACH SIDE MUST NOT BE MORE THAN 20 INCHES. NO UNREPAIRED DAMAGE IS PERMITTED TO THE ADJACENT ALUMINUM STRUCTURE. ALL DAMAGE MUST BE REPAIRED AS GIVEN IN THE APPLICABLE SRM INSTRUCTIONS.

2 CRACKS THAT TERMINATE AT THE INITIAL FASTENER HOLES DO NOT NEED TO BE STOP DRILLED.

VAPOR BARRIER

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

AFT VIEW FACING FORWARD

B

436084 S0000141442_V3

**Wing Center Section Rear Spar Allowable Damage**
**Figure 101 (Sheet 2 of 2)**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Do the steps that follow for the parts made of aluminum.

(1) Remove the damage as necessary.

(a) Refer to 51-10-02 for the inspection and removal of damage.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(a) Flap peen or shot peen the reworked areas of the spar chords or the upper web.

1) Refer to 51-20-06 for shot peen intensity and shot number.

2) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

(b) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(c) Apply a layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-41-02.

C. Do the steps that follow for the lower left or right webs made of Glass Fabric Reinforced Plastic (GFRP).

(1) Refer to 51-30-05 for the tools.

(2) Do the steps that follow to clean the surface.

**WARNING:** DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(a) Remove loose dirt, unwanted grease, or oil. Be careful not to spread the dirt over a larger area than necessary. Refer to SOPM 20-30-99 for the applicable solvents.

(b) Wet the surface with solvent and wipe or scrub with a wiper, sponge, cheesecloth or brush. Repeat this step as necessary to remove the dirt.

(c) Rinse the surface with fresh solvent and wipe with a clean sponge or cheesecloth.

(d) Wipe off the excess solvent and let the surface dry.

(e) Abrade the surface.

**WARNING:** DO NOT BREATHE THE DUST THAT IS MADE WHEN YOU ABRADE OR CUT CURED COMPOSITE MATERIALS, OR LET THE DUST GET IN YOUR EYES OR ON YOUR SKIN. USE EYE PROTECTION AND PROTECTIVE CLOTHING. MAKE SURE THAT YOU USE RESPIRATORY EQUIPMENT WHEN YOU WORK IN CLOSED LOCATIONS. USE A VACUUM TABLE OR PORTABLE VACUUM TO REMOVE THE DUST WHILE YOU ABRADE OR CUT. IF YOU DO NOT OBEY, INJURY TO PERSONS CAN OCCUR.

(WARNING PRECEDES)

**CAUTION**: DO NOT DAMAGE THE FIBERS WHEN YOU MAKE THE SURFACES ROUGH WITH ABRASIVE PAPER. IF YOU DO NOT OBEY, YOU WILL CAUSE A DECREASE IN THE STRENGTH OF THE PART.

1) Protect the undamaged area of the web from the abrasive material. Mask around the damage area as necessary. Use material that is resistant to the abrasive material, such as vinyl tape, or Teflon tape.

2) Abrade (sand) the surface with one of the steps that follow to remove the coating.

**NOTE**: If you use an orbital driven sander, set the sander below 20,000 revolutions per minute (RPM).

a) Sand the surface with 80 grit or finer abrasive paper for extensive sanding.

b) Sand the surface with 150 grit or finer abrasive paper for final sanding.

c) Finish the surface with 1/0 or finer garnet paper or cloth or fine/ultrafine abrasive fabric pad (Scotch-Brite).

(f) Repeat Paragraph 2.C.(2)(a) through 2.C.(2)(d) to re-clean the repair area.

(3) Do one of the steps that follow to seal the damage that is not more than one ply deep and that agrees with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 2.

(a) Make an interim seal on the side common to the damage.

1) Refer to Paragraph 2.C.(2)/ALLOWABLE DAMAGE 2 for the clean up procedure.

**NOTE**: Remove contamination from an area 1.50 inches (38 mm) minimum in all directions of the crack.

2) Apply aluminum foil tape (speed tape).

3) Keep a record of the location.

4) Make sure the tape is in satisfactory condition at each 600 flight cycles interval, or more frequently.

(b) Make a permanent seal.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to SOPM 20-44-04.

3) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Do one of the steps that follow to seal the cracks that are not more than 16.0 inches (406.4 mm) long and that are shown in zone 1 as given in Figure 101 (Sheet 2). The cracks are Category B. Refer to 51-00-06, GENERAL for the definitions of the different categories.

(a) Make an interim seal on the wheel well side of the web only.

1) Perform a 0.25 inch diameter stop drill at the crack ends. Refer to 51-10-02, GENERAL for the stop drill procedures.

2) Use vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04, REPAIR GENERAL.

3) Refer to Paragraph 2.C.(2)/ALLOWABLE DAMAGE 2 for the clean up procedure.

**NOTE**: Remove contamination from an area 1.50 inches (38 mm) minimum in all directions of the crack.

4) Apply aluminum foil tape (speed tape).
5) Keep a record of the crack length and the location.
6) Do a visual inspection of the repair area at 600 flight cycle intervals or more frequently.
   a) Make sure that crack has not grown past the interim seal.
   b) If the crack has grown past the interim seal, refer to Paragraph 4./ALLOWABLE DAMAGE 2 for allowable damage limits.
   c) If the total crack length meets the allowable damage limits in Paragraph 4./ALLOWABLE DAMAGE 2, refer to Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2 for repair instructions.
   d) If the total crack length is more than the allowable limits given in Paragraph 4./ALLOWABLE DAMAGE 2, do a permanent repair as given in 51-70-04, REPAIR 10 to remove the crack. As an alternative, you can repair part of the crack so that the crack length is less than or equal to the maximum allowable damage limit. Apply the interim seal over the crack that is left as given in Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2.

(b) Make an interim seal on the wheel well side of the web only.

1) Perform a 0.25 inch diameter stop drill at the crack ends. Refer to 51-10-02, GENERAL for the stop drill procedures.
2) Use vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04, REPAIR GENERAL.
3) Refer to Paragraph 2.C.(2)/ALLOWABLE DAMAGE 2 for the clean up procedure.

**NOTE**: Remove contamination from an area 1.50 inches (38 mm) minimum in all directions of the crack.

4) Apply BMS 5-45 or BMS 5-95 sealant (class and type optional) plus one ply of BMS 9-3, style 1581 fiberglass cloth on the cracked areas as follows.
   a) Cut a piece of fiberglass cloth to fit 1.00/1.50 inches (25/38 mm) in all directions beyond the crack
   b) Layout the fiberglass cloth on a clean work surface.
   c) Impregnate the cloth with BMS 5-45 or BMS 5-95 sealant.
   d) Apply sealant .03/.10 inch (0.76/2.5 mm) thick X 1.00/1.50 inches (25/38 mm) in all directions over the crack.
   e) Apply and press the impregnated fiberglass cloth lightly on vapor barrier. Make sure you have sealant around the edge of the fiberglass cloth.
   f) Apply an additional .03/.10 inch (0.76/2.5 mm) thick layer of sealant over the cloth. Make sure the sealant completely covers the fiberglass cloth. Refer to 51-20-05, GENERAL.
5) Finish with BMS 10-79, Type 3 or BMS 10-103, Type 1 primer and BMS 10-60 enamel.
6) Keep a record of the crack length and the repair location.
7) Do a visual inspection of the repair area at 600 flight cycle intervals or more frequently.

a) Make sure that crack has not grown past the interim seal.

b) If the crack has grown past the interim seal, refer to Paragraph 4./ALLOWABLE DAMAGE 2 for allowable damage limits.

c) If the total crack length meets the allowable damage limits in Paragraph 4./ALLOWABLE DAMAGE 2, refer to Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2 for repair instructions.

d) If the total crack length is more than the allowable limits given in Paragraph 4./ALLOWABLE DAMAGE 2, do a permanent repair as given in 51-70-04, REPAIR 10 to remove the crack. As an alternative, you can repair part of the crack so that the crack length is less than or equal to the maximum allowable damage limit. Apply the interim seal over the crack that is left as given in Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2.

## 3. References

| Reference | Title |
|---|---|
| 51-00-06, GENERAL | Structural Repair Definitions |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-05, GENERAL | Repair Sealing |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-70-04, REPAIR 10 | Repair of Damage That is More Than 0.50 Inch in Solid Laminates |
| 51-70-04, REPAIR GENERAL | Repair Procedures for Wet Layup Materials |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Chords (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , and F .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Upper Web (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , and G .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , E , F , and G.

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Detail H .

(4) Holes and Punctures are not permitted.

C. Lower Center Web (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , and G.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , E , F and G.

(3) Dents:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Detail H.

(4) Holes and Punctures:

(a) Damage is permitted if:

1) The damage is a maximum of 0.25 inch diameter after the damage is removed.

2) The damage is a minimum of 0.5 inch from all padups.

3) The damage is a minimum of 1.00 inch from any hole, edge of part, or other damage.

4) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet.

a) Install the rivet wet with BMS 5-26 sealant.

D. Lower Left Web and Lower Right Web (GFRP)

**NOTE**: Refer to Definitions of the Damage Size, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Cracks:

(a) The damage is permitted if they are:

1) Less than or equal to the dimensions given in Paragraph 2.C. and as shown in Figure 101, Sheet 2.

2) Sealed as given in Paragraph 2.C.

(2) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(a) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

1) A maximum of one ply in depth.

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply.

(b) A maximum of 5.0 inches in length.

(c) A maximum of 0.25 inch in width.

(d) A minimum of 0.50 inch away from the edge of a fastener hole.

(e) A minimum of 0.50 inch away from the edge of other damage. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet installed wet with BMS 5-26 sealant.

(5) Delaminations are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) Are sealed as given in Paragraph 2.

E. Stiffeners (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details , A , B , D and I.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F and I.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

F. Support Fitting (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , and B.

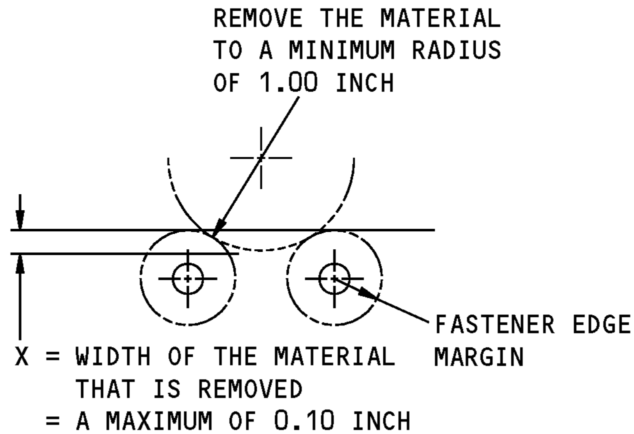
(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the edge damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , E , and J .

(b) Bushing damage is not permitted.

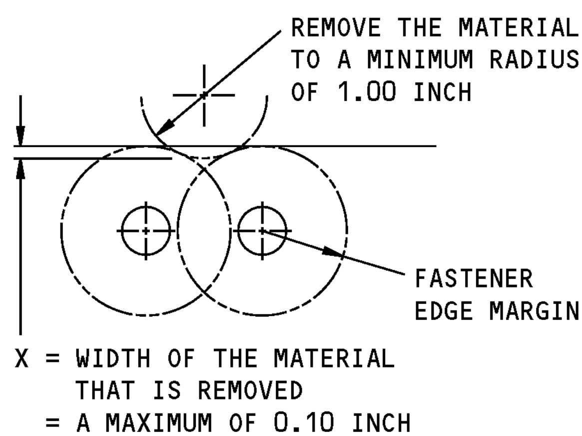
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.




REMOVAL OF DAMAGED MATERIAL AT
EDGES WHERE THE FASTENER EDGE
MARGINS DO NOT HAVE AN OVERLAP

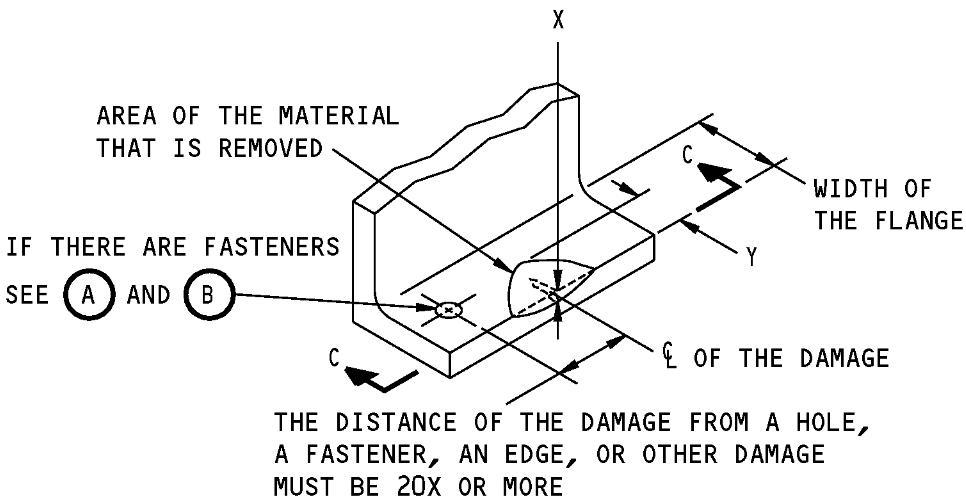
A




REMOVAL OF DAMAGED MATERIAL AT
EDGES WHERE THE FASTENER EDGE
MARGINS HAVE AN OVERLAP

B

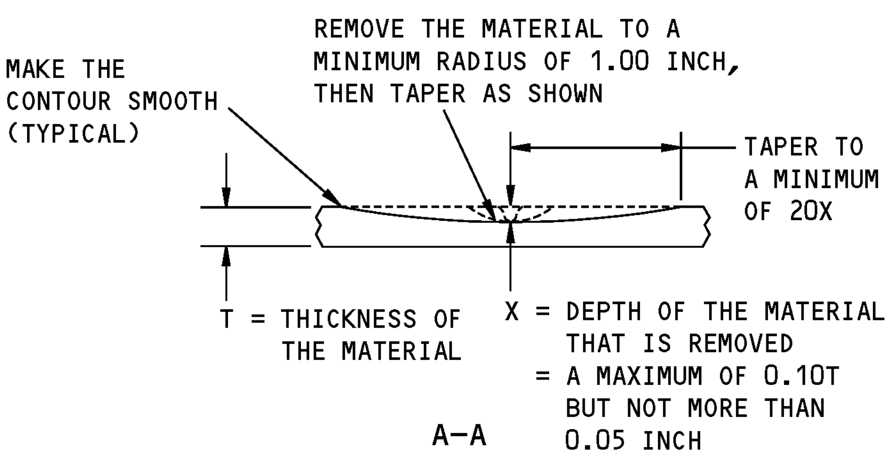
**Allowable Damage Limits**
**Figure 102 (Sheet 1 of 6)**




Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

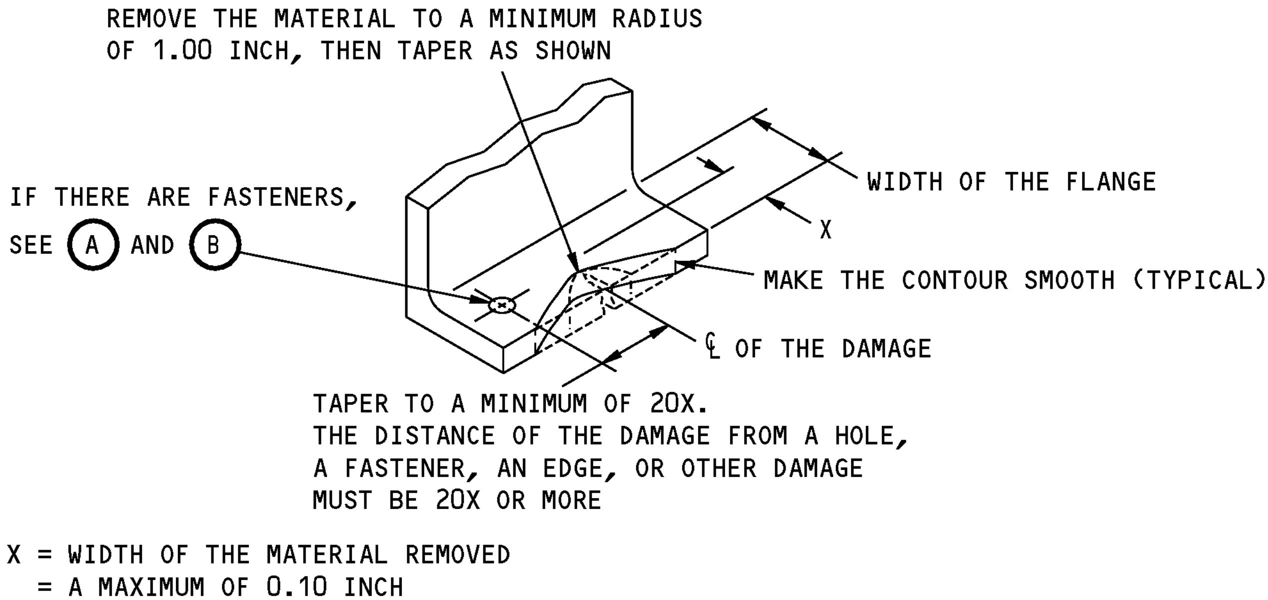
REMOVAL OF DAMAGED MATERIAL ON A
SURFACE AT AN EDGE OF AN EXTRUDED PART

C



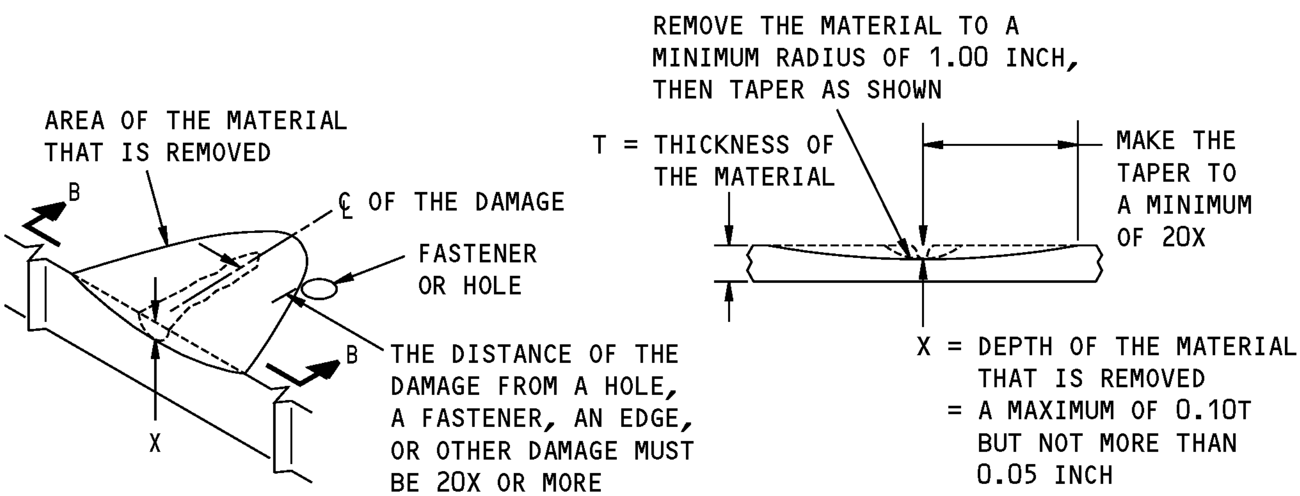

A-A

**Allowable Damage Limits**
**Figure 102 (Sheet 2 of 6)**




REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF AN EXTRUDED PART

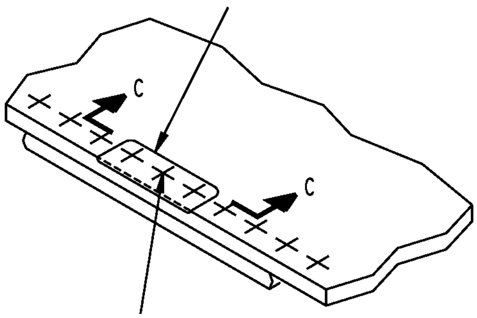
D




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

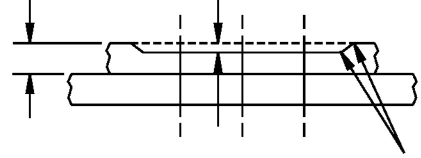
E

**Allowable Damage Limits**
**Figure 102 (Sheet 3 of 6)**

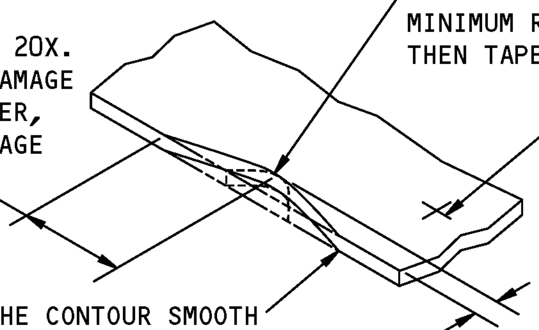



REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

F




C-C

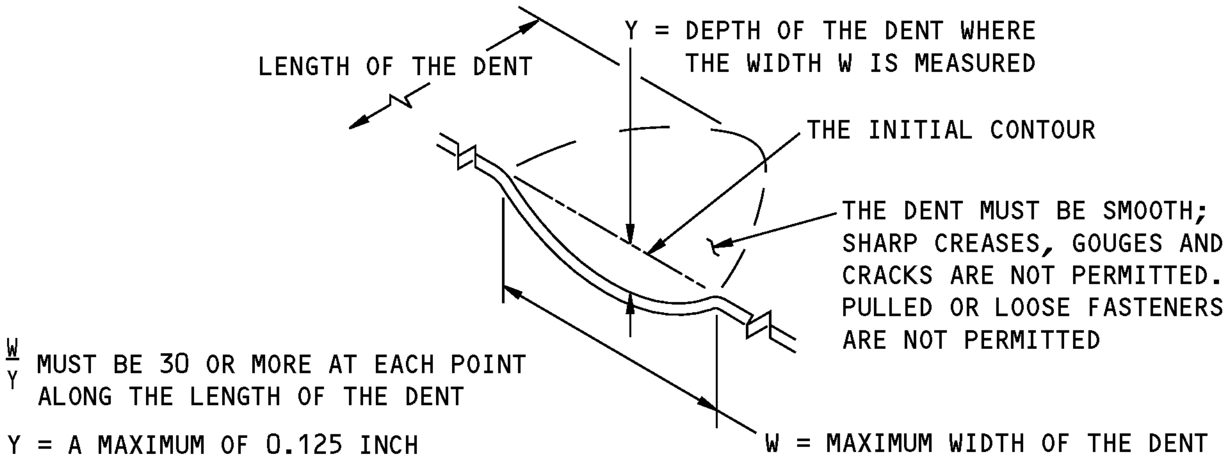



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

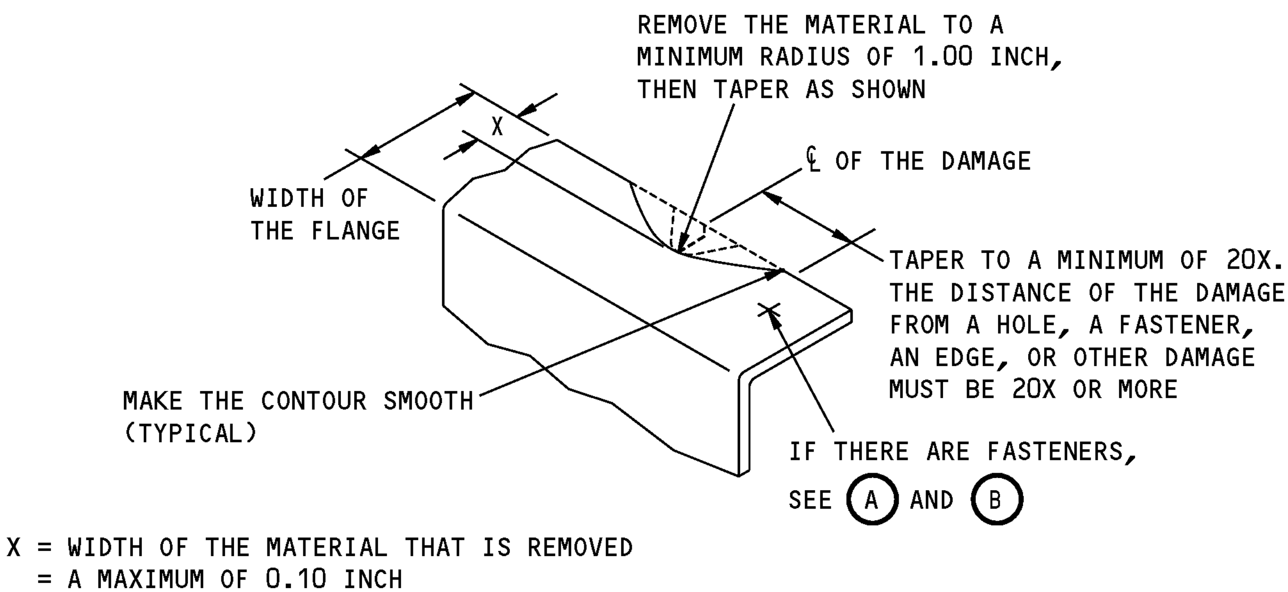
G

**Allowable Damage Limits**
**Figure 102 (Sheet 4 of 6)**




DENT THAT IS PERMITTED

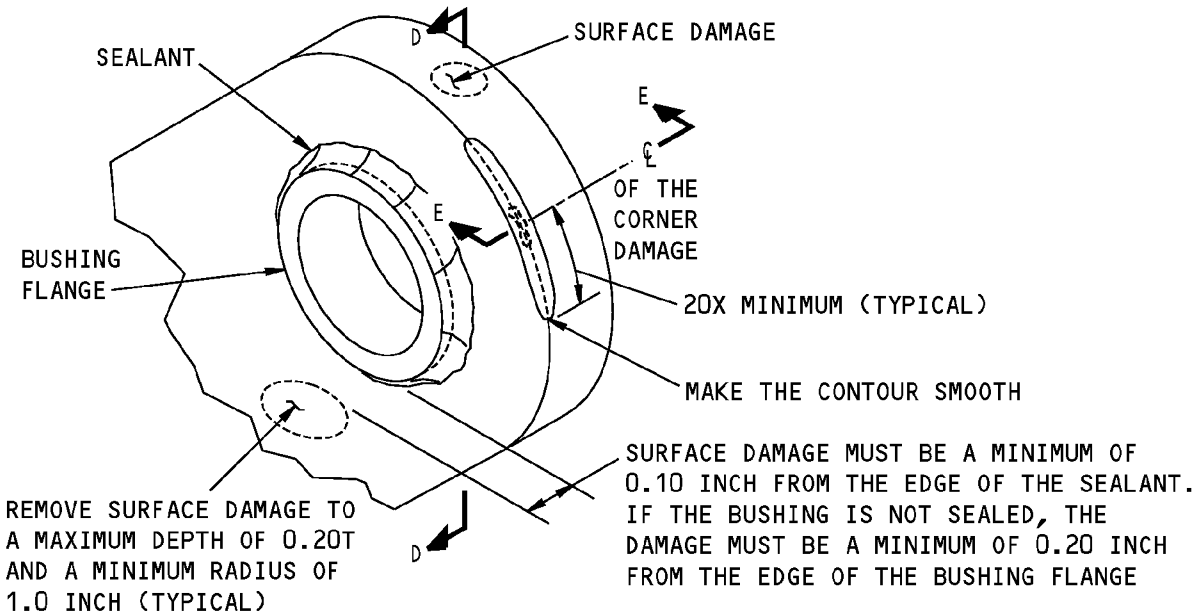
H




REMOVAL OF DAMAGED MATERIAL ON AN EDGE OF A FORMED PART

I

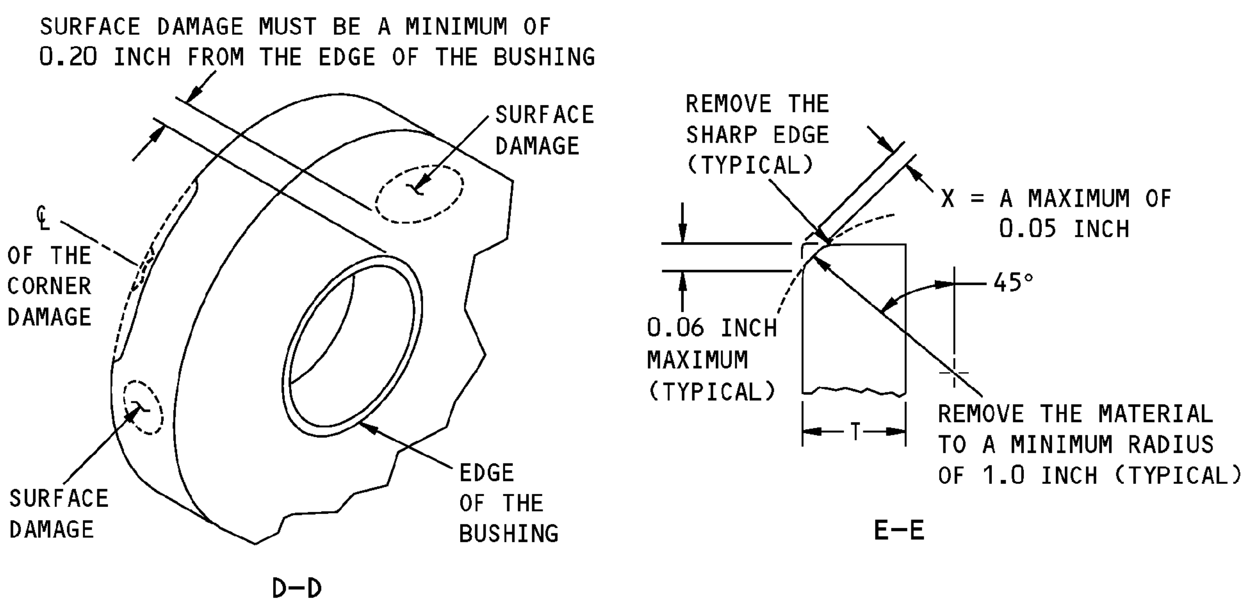
**Allowable Damage Limits**
**Figure 102 (Sheet 5 of 6)**



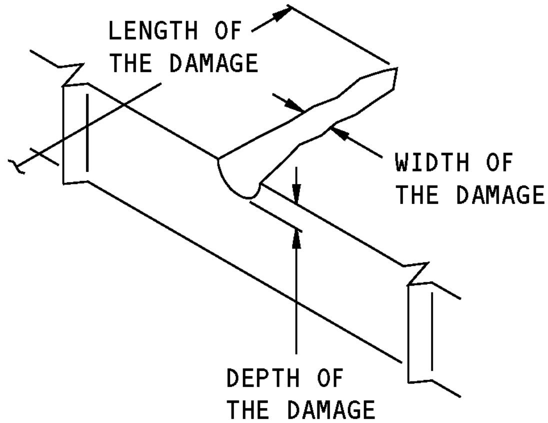

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

J

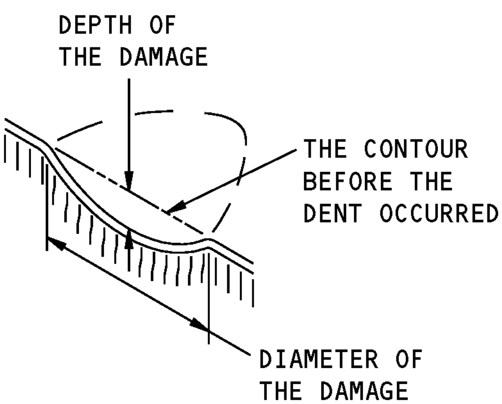



**Allowable Damage Limits**
**Figure 102 (Sheet 6 of 6)**



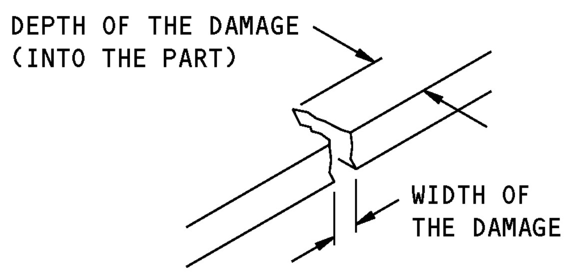

SIZE DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A




SIZE DEFINITIONS FOR
DENT DAMAGE

B

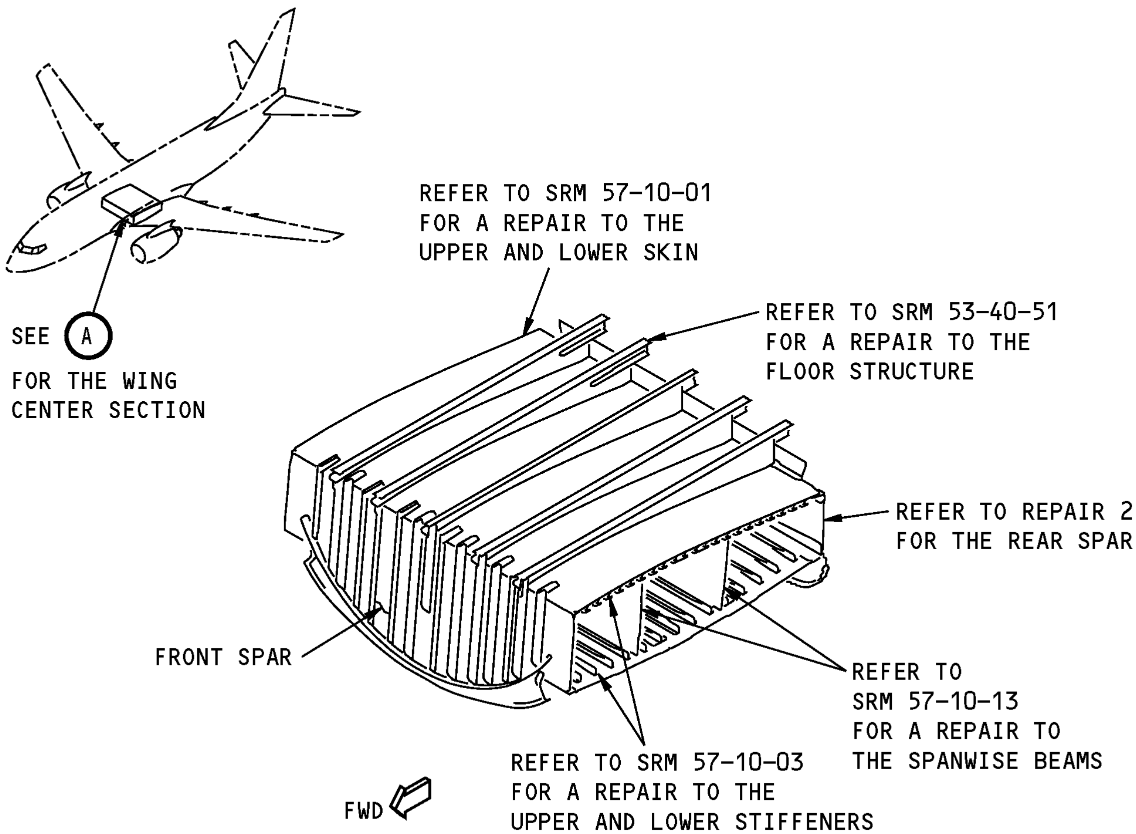



SIZE DEFINITIONS FOR
EDGE DAMAGE

C

**Definitions of the Damage Size**
**Figure 103**

## REPAIR 1 - WING CENTER SECTION FRONT SPAR




NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

A

**Wing Center Section Front Spar Repair**
**Figure 201**

## REPAIR 2 - WING CENTER SECTION REAR SPAR

1. **Applicability**

   A. This repair is applicable to damage to the wing center section rear spar fiberglass vapor barrier as shown in Figure 201.

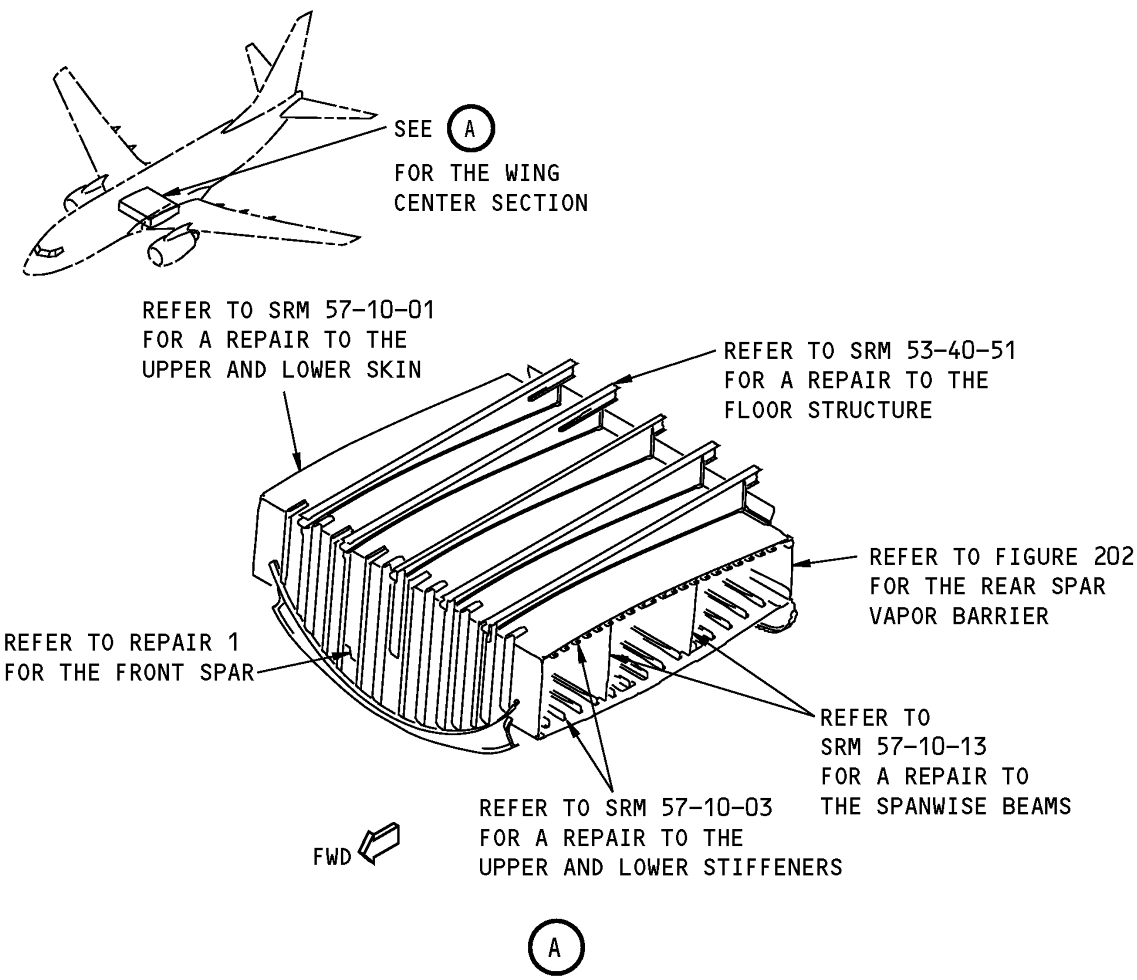
   B. This repair is applicable to airplanes with line numbers 1 through 1639 only. For airplanes with line numbers 1640 and on, contact The Boeing Company for alternative repair instructions, or you can use the wet layup repair as given in 51-70-04, REPAIR 10.

   C. As an alternative to this repair, you can use the wet layup repairs as given in 51-70-04, REPAIR 10. The alternative wet lay-up repair as given in 51-70-04, REPAIR 10 is applicable to all line numbers.

2. **General**

   A. This repair is a Category A repair. Refer to 51-00-06, GENERAL to find the different categories of repairs.

   B. Figure 204 addresses repair action to be taken for cracks in the rear spar vapor barrier web on an individual bay by bay basis. Cracks inside any individual bay which do not extend closer than 2.0 inches to adjacent structure require repair only to the affected bay. Cracks in an individual bay which are closer than 2.0 inches to an adjacent bay also require repair to the adjacent bay.




**Wing Center Section Rear Spar Repair**
**Figure 201**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06, GENERAL | Structural Repair Definitions |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01, GENERAL | Protective Treatment of Metallic and Composite Materials |
| 51-20-05, GENERAL | Repair Sealing |
| 51-70-04, REPAIR 10 | Repair of Damage That is More Than 0.50 Inch in Solid Laminates |
| AMM 28-11-00 | FUEL TANKS |
| AMM 51-31-00 | SEALS AND SEALING |
| 737 NDT Part 6, 51-00-00, Figure 4 | Surface Inspection of Aluminum Parts |

**4. Repair Instructions For The 3 Bay Doubler Repair As Given In Figure 203**

A. Get access to the repair area.

B. If there is a crack, then drill a 0.25 inch (6.35 mm) diameter stop hole at the crack ends as given in 51-10-02, GENERAL.

**NOTE**: A stop hole is not necessary if the crack ends at a fastener hole.

C. Fabricate the repair parts as shown in Figure 203 and as given in Table 201.

(1) Make the surface finish to all cut surfaces 125 microinches Ra, or smoother.

(2) Make the internal corner radius 0.50 inch (12.70 mm) minimum. Make the external corner radius 0.25 inch (6.35 mm) minimum.

D. Assemble the repair parts and drill the fastener holes as shown in Figure 203.

**Table 201**:

| REPAIR MATERIAL FOR FIGURE 203 | | | |
|---|---|---|---|
| **PART NUMBER** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Repair Doubler | 1 | Use 2024-T3 clad sheet that is 0.040 inch (1.02 mm) thick |
| [2] | Filler | 3 | Use 2024-T3 clad sheet with thickness as necessary to limit any gaps to a maximum of 0.005 inch (0.13 mm). |

E. Do a penetrant inspection of the joggle locations to make sure that there are no cracks.

F. Remove the repair parts.

G. Remove all nicks, scratches, burrs, and sharp edges from the repair parts.

H. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the vapor barrier. Refer to 51-20-01, GENERAL.

I. Apply one layer of BMS 10-11, Type I primer to the repair parts and to the bare surfaces of the vapor barrier. Refer to SOPM 20-41-02.

J. Install the repair parts wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL. Apply a fillet seal of BMS 5-95 sealant to the edges of the part [1] repair doubler.

K. Install the fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

**NOTE**: It is permitted to use 1/64 inch diameter or 1/32 inch diameter oversize fasteners, if necessary. Make sure there is a minimum of 1.5D (D = fastener diameter) edge margin.

(1) Install the fasteners with the fastener head on the opposite side of the part [1] repair doubler.

**5. Repair Instructions For The Full Cross Brace Repair As Given In Figure 204**

A. Get access to the repair area.

B. If there is a crack, then drill a 0.25 inch (6.35 mm) diameter stop hole at the crack ends as given in 51-10-02, GENERAL.

**NOTE**: A stop hole is not necessary if the crack ends at a fastener hole.

C. Fabricate the repair parts as shown in Figure 204 and as given in Tables 202 through 214.

(1) Make the surface finish to all cut surfaces 125 microinches Ra, or smoother.

(2) Follow all radius callouts as described in Figure 204.

D. Assemble the repair parts and drill the fastener holes as shown in Figure 204.

Table 202:

| REPAIR MATERIAL FOR BAY 1L AND BAY 1R | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -1 | -2 | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 34.4 inches |
| [2] | -3 | -4 | Crossbrace | 1 | |
| [3] | -5 | -6 | Crossbrace | 1 | |
| [4] | -7 | -8 | Joint Angle | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches |
| [5] | -9 | -10 | Joint Angle | 1 | |
| [6] | -11 | -12 | Strap | 1 | Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 34.4 inches |
| [7] | -13 | -14 | Strap | 1 | |
| [8] | -15 | -16 | Strap | 1 | |
| [9] | -17 | -18 | Strap | 1 | Use 7075-T6 bare sheet that is 0.036 inch thick X 4.0 inches X 34.4 inches |
| [10] | -19 | -20 | Strap | 1 | |
| [11] | -21 | -22 | Strap | 1 | |
| [12] | -23 | -24 | Strap | 1 | Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 34.4 inches |
| [13] | -25 | -26 | Strap | 1 | |
| [14] | -27 | -28 | Strap | 1 | |

*[1] These part numbers are for reference only.

Table 203:

| REPAIR MATERIAL FOR BAY 2L AND BAY 2R | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -29 | -30 | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 20.1 inches |
| [2] | -31 | -32 | | | |
| [3] | -33 | -34 | | | |
| [4] | -35 | -36 | Joint Angle | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches |
| [5] | -39 | -40 | | | |
| [6] | -41 | -42 | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch X 4.0 inches X 20.1 inches |
| [7] | -43 | -44 | | | |
| [8] | -45 | -46 | | | |

*[1] These part numbers are for reference only.

Table 204:

| REPAIR MATERIAL FOR BAY 3L AND BAY 3R | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -47 | -48 | Crossbrace | 1 | Use 7075-T6 in BAC1503-100146 extruded angle X 8.5 inches |
| [2] | -49 | -50 | Crossbrace | 1 | Use 7075-T6 in AND10134-1403 extruded angle X 18.5 inches |
| [3] | -51 | -52 | Crossbrace | 1 | Use 7075-T6 in BAC1503-100146 extruded angle X 10.5 inches |
| [4] | -53 | -54 | Joint Angle | 1 | Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches |
| [5] | -55 | -56 | Joint Angle | 1 | |
| [6] | -57 | -58 | Strap | 1 | Use 7075-T6 bare sheet that is 0.063 inch X 4.0 inches X 33.2 inches |
| [7] | -59 | -60 | Strap | 1 | |
| [8] | -61 | -62 | Strap | 1 | |
| [9] | -63 | -64 | Strap | 1 | Use 7075-T6 bare sheet that is 0.036 inch thick X 4.0 inches X 33.2 inches |
| [10] | -65 | -66 | Strap | 1 | |
| [11] | -67 | -68 | Strap | 1 | |
| [12] | -69 | -70 | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 33.2 inches. |
| [13] | -71 | -72 | Strap | 1 | |
| [14] | -73 | -74 | Strap | 1 | |

*[1] These part numbers are for reference only.

Table 205:

| REPAIR MATERIAL FOR BAY 4L AND BAY 4R | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -75 | -76 | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 29.0 inches |
| [2] | -77 | -78 | Crossbrace | 1 | |
| [3] | -81 | -82 | Crossbrace | 1 | |
| [4] | -83 | -84 | Strap | 1 | Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 29.0 inches |
| [5] | -85 | -86 | Strap | 1 | |
| [6] | -87 | -88 | Strap | 1 | |

*[1] These part numbers are for reference only.

Table 206:

| REPAIR MATERIAL FOR BAY 5L | | | | | |
|---|---|---|---|---|---|
| | PART NUMBER 111A1152 *[1] | | | | |
| ITEM | LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -89 | - | Crossbrace | 1 | Use 7075-T6 in AND10134-1403 extruded angle X 5.0 inches |
| [2] | -91 | - | Crossbrace | 1 | Use 7075-T6 in AND10134-1403 extruded angle X 11.5 inches |
| [3] | -93 | - | Crossbrace | 1 | Use 7075-T6 in BAC1503-100146 extruded angle X 5.0 inches |
| [4] | -95 | - | Joint Angle | 1 | Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches |
| [5] | -97 | - | Joint Angle | 1 | |
| [6] | -99 | - | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches width X 21.6 inches length |
| [7] | -101 | - | Strap | 1 | |
| [8] | -103 | - | Strap | 1 | |
| [9] | -105 | - | Strap | 1 | |
| [10] | -107 | - | Strap | 1 | |
| [11] | -109 | - | Strap | 1 | |

*[1] These part numbers are for reference only.

Table 207:

| REPAIR MATERIAL FOR BAY 5R | | | | | |
|---|---|---|---|---|---|
| | PART NUMBER 111A1152 *[1] | | | | |
| ITEM | LH | RH | PART | QUANTITY | MATERIAL |
| [1] | - | -181 | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 16.2 inches |
| [2] | - | -183 | Crossbrace | 1 | |
| [3] | - | -185 | Strap | 1 | Use7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 16.2 inches |
| [4] | - | -187 | Strap | 1 | |
| [5] | - | -189 | Strap | 1 | |
| [6] | - | -191 | Strap | 1 | Use 7075-T6 bare sheet that is 0.036 inch thick X 4.0 inches X 16.2 inches |

*[1] These part numbers are for reference only.

Table 208:

| REPAIR MATERIAL FOR BAY 6L AND BAY 6R | | | | | |
|---|---|---|---|---|---|
| | PART NUMBER 111A1152 *[1] | | | | |
| ITEM | LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -111 | -112 | Crossbrace | 1 | Use 7075-T6 in AND10134-2001 extruded angle X 24.0 inches |
| [2] | -113 | -114 | Crossbrace | 1 | |
| [3] | -115 | -116 | Crossbrace | 1 | |

| REPAIR MATERIAL FOR BAY 6L AND BAY 6R | | | | | |
|---|---|---|---|---|---|
| | PART NUMBER 111A1152 *[1] | | | | |
| ITEM | LH | RH | PART | QUANTITY | MATERIAL |
| [4] | -117 | -118 | Joint Angle | 1 | Use 7075-T6 AND10134-2001 extruded angle X 6.0 inches |
| [5] | -119 | -120 | Joint Angle | 1 | |
| [6] | -121 | -122 | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 24.0 inches |
| [7] | -123 | -124 | Strap | 1 | |
| [8] | -125 | -126 | Strap | 1 | |

*[1] These part numbers are for reference only.

**Table 209**:

| REPAIR MATERIAL FOR BAY 7L AND BAY 7R | | | | | |
|---|---|---|---|---|---|
| | PART NUMBER 111A1152 *[1] | | | | |
| ITEM | LH | LH | PART | QUANTITY | MATERIAL |
| [1] | -127 | -128 | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 27.7 inches |
| [2] | -129 | -130 | Crossbrace | 1 | |
| [3] | -131 | -132 | Crossbrace | 1 | |
| [4] | -133 | -134 | Joint Angle | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches |
| [5] | -135 | -136 | Joint Angle | 1 | |
| [6] | -137 | -138 | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 27.7 inches |
| [7] | -139 | -140 | Strap | 1 | |
| [8] | -141 | -142 | Strap | 1 | |
| [9] | -249 | -250 | Strap | 1 | |
| [10] | -251 | -252 | Strap | 1 | |
| [11] | -253 | -254 | Strap | 1 | |
| [12] | -143 | -144 | Bracket | 2 | Use 7075-T6 AND10134-2001 extruded angle X 6.0 inches |

*[1] These part numbers are for reference only.

**Table 210**:

| REPAIR MATERIAL FOR BAY 8L | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] | | PART | QUANTITY | MATERIAL |
| | LH | RH | | | |
| [1] | -145 | - | Crossbrace | 1 | Use 7075-T6 in AND10134-1403 extruded angle X 22.0 inches |
| [2] | -147 | - | Crossbrace | 1 | |
| [3] | -149 | - | Crossbrace | 1 | |
| [4] | -151 | - | Joint Angle | 1 | Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches |
| [5] | -153 | - | Joint Angle | 1 | |

| REPAIR MATERIAL FOR BAY 8L | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] | | PART | QUANTITY | MATERIAL |
| | LH | RH | | | |
| [6] | -155 | - | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 22.0 inches |
| [7] | -157 | - | Strap | 1 | |
| [8] | -159 | - | Strap | 1 | |

*[1] These part numbers are for reference only.

Table 211:

| REPAIR MATERIAL FOR BAY 8R | | | | | |
|---|---|---|---|---|---|
| | PART NUMBER 111A1152 *[1] | | | | |
| ITEM | LH | RH | PART | QUANTITY | MATERIAL |
| [1] | - | -193 | Crossbrace | 1 | Use 7075-T6 in AND10134-1403 extruded angle X 32.2 inches |
| [2] | - | -195 | Crossbrace | 1 | |
| [3] | - | -197 | Crossbrace | 1 | |
| [4] | - | -199 | Crossbrace | 1 | |
| [5] | - | -201 | Crossbrace | 1 | |
| [6] | - | -231 | Crossbrace | 1 | |
| [7] | - | -203 | Joint Angle | 1 | Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches |
| [8] | - | -205 | Joint Angle | 1 | |
| [9] | - | -207 | Joint Angle | 1 | |
| [10] | - | -209 | Joint Angle | 1 | |
| [11] | - | -211 | Strap | 1 | Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 32.2 inches |
| [12] | - | -213 | Strap | 1 | |
| [13] | - | -215 | Strap | 1 | |
| [14] | - | -217 | Strap | 1 | |
| [15] | - | -219 | Strap | 1 | |
| [16] | - | -221 | Strap | 1 | |
| [17] | - | -223 | Strap | 1 | Use 7075-T6 bare sheet that is 0.125 inch thick X 4.0 inches X 32.2 inches |
| [18] | - | -225 | Strap | 1 | |
| [19] | - | -227 | Strap | 1 | |
| [20] | - | -233 | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 32.2 inches |
| [21] | - | -235 | Strap | 1 | |
| [22] | - | -237 | Strap | 1 | |
| [23] | - | -239 | Strap | 1 | Use 7075-T6 bare sheet that is 0.025 inch thick X 4.0 inches X 32.2 inches |
| [24] | - | -241 | Strap | 1 | |
| [25] | - | -243 | Strap | 1 | |

*[1] These part numbers are for reference only.

Table 212:

| REPAIR MATERIAL FOR BAY 9L AND BAY 9R | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -161 | -162 | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 22.6 inches |
| [2] | -163 | -164 | Crossbrace | 1 | |
| [3] | -165 | -166 | Crossbrace | 1 | |
| [4] | -167 | -168 | Joint Angle | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches |
| [5] | -169 | -170 | Joint Angle | 1 | |
| [6] | -171 | -172 | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 22.6 inches |
| [7] | -173 | -174 | Strap | 1 | |
| [8] | -175 | -176 | Strap | 1 | |

*[1] These part numbers are for reference only.

Table 213:

| REPAIR MATERIAL FOR BAY 10L | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] LH | RH | PART | QUANTITY | MATERIAL |
| [1] | -177 | - | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 22.0 inches |
| [2] | -179 | - | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 22.0 inches |

*[1] These part numbers are for reference only.

Table 214:

| REPAIR MATERIAL FOR BAY 10R | | | | | |
|---|---|---|---|---|---|
| ITEM | PART NUMBER 111A1152 *[1] LH | RH | PART | QUANTITY | MATERIAL |
| [1] | - | -245 | Crossbrace | 1 | Use 7075-T6 in AND10134-1401 extruded angle X 15.7 inches |
| [2] | - | -247 | Strap | 1 | Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 15.7 inches. |

*[1] These part numbers are for reference only.

E. Remove the repair parts.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and the metal parts of initial structure.

G. Apply a chemical conversion coating to the repair parts and the metal parts of initial structure. Refer to 51-20-01, GENERAL.

H. Apply one layer of BMS 10-11, Type I primer to the repair parts and the metal parts of initial structure. Refer to SOPM 20-41-02.

I. Clean the vapor barrier web assembly.

**WARNING:** DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(1) Remove loose dirt, unwanted grease, or oil. Be careful not to spread the dirt over a larger area than necessary. Refer to SOPM 20-30-99 for the applicable solvents.

(2) Wet the surface with solvent and wipe or scrub with a wiper, sponge, cheesecloth or brush. Repeat this step as necessary to remove the dirt.

(3) Rinse the surface with fresh solvent and wipe with a clean sponge or cheesecloth.

(4) Wipe off the excess solvent and let the surface dry.

(5) Abrade the surface.

**WARNING:** DO NOT BREATHE THE DUST THAT IS MADE WHEN YOU ABRADE OR CUT CURED COMPOSITE MATERIALS, OR LET THE DUST GET IN YOUR EYES OR ON YOUR SKIN. USE EYE PROTECTION AND PROTECTIVE CLOTHING. MAKE SURE THAT YOU USE RESPIRATORY EQUIPMENT WHEN YOU WORK IN CLOSED LOCATIONS. USE A VACUUM TABLE OR PORTABLE VACUUM TO REMOVE THE DUST WHILE YOU ABRADE OR CUT. IF YOU DO NOT OBEY, INJURY TO PERSONS CAN OCCUR.

**CAUTION:** DO NOT DAMAGE THE FIBERS WHEN YOU MAKE THE SURFACES ROUGH WITH ABRASIVE PAPER. IF YOU DO NOT OBEY, YOU WILL CAUSE A DECREASE IN THE STRENGTH OF THE PART.

(a) Protect the undamaged area of the web from the abrasive material. Mask around the crack a minimum of 2 inches from the crack. Use material that is resistant to the abrasive material, such as vinyl tape, or Teflon tape.

(b) Abrade (sand) the surface with one of the steps that follow to remove the coating.

**NOTE**: If you use an orbital driven sander, set the sander below 20,000 revolutions per minute (RPM).

1) Sand the surface with 80 grit or finer abrasive paper for extensive sanding.

2) Sand the surface with 150 grit or finer abrasive paper for final sanding.

3) Finish the surface with 1/0 or finer garnet paper or cloth or fine/ultrafine abrasive fabric pad (Scotch-Brite).

(6) Repeat steps (1) through (4) above to re-clean the repair area.

J. Install the repair parts with BMS 5-45 or BMS 5-95 sealant between the mating surfaces. Refer to Figure 204.

K. Install the fasteners with the fastener head on the opposite side of the repair parts.

**NOTE**: It is permitted to use 1/64 inch diameter or 1/32 inch diameter oversize fasteners, if necessary. Make sure there is a minimum of 2.0D (D = fastener diameter) edge/end margin.

L. Apply a fillet of BMS 5-45 or BMS 5-95 sealant to all part edges as given in the Figure 204 instructions. Refer to 51-20-05, GENERAL and AMM SUBJECT 51-31-00.

M. Apply a fillet of BMS 5-45 or BMS 5-95 sealant to all fasteners as given in the Figure 204 instructions. Refer to 51-20-05, GENERAL and AMM SUBJECT 51-31-00.

N. Fill the stop drilled hole with BMS 5-45 or BMS 5-95 sealant flush to -0.00/+0.03 inch. Refer to 51-20-05, GENERAL and AMM SUBJECT 51-31-00.

O. Apply BMS 5-45 or BMS5-95 (class and type optional) and one ply of BMS 9-3,style 1581 fiberglass cloth over the part of the crack that you can see. Use the steps that follow:

(1) Cut a piece of BMS 9-3, style 1581 fiberglass cloth to fit 1.00/1.50 inches (25/38 mm) in all directions beyond the crack.

(2) Layout the fiberglass cloth on a clean work surface.

(3) Impregnate the cloth with BMS 5-45 or BMS5-95 sealant.

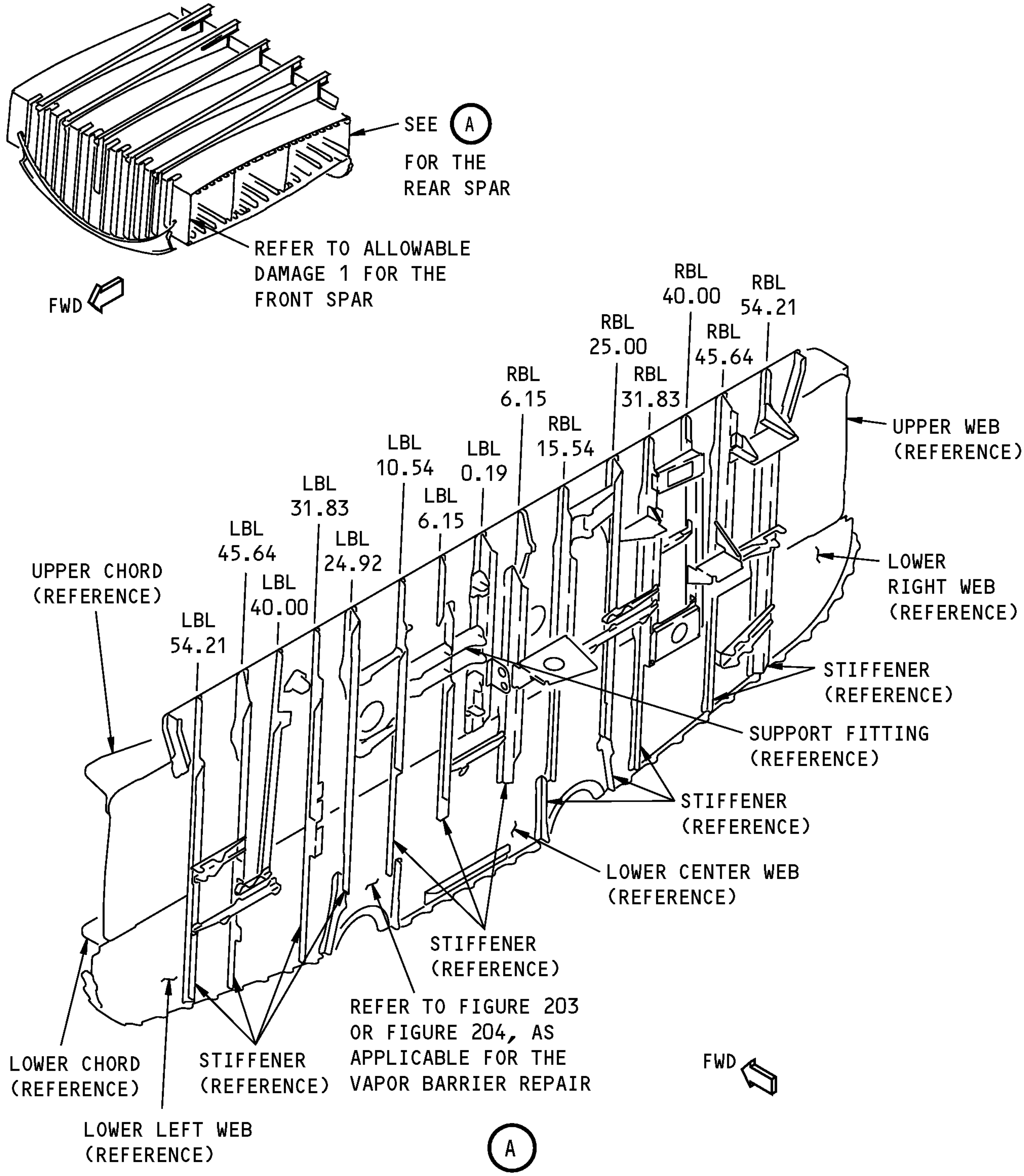
(4) Apply sealant .03/.10 inch (0.76/2.5 mm) thick X 1.00/1.50 inches (25/38 mm) in all directions over the crack.

(5) Apply and press the impregnated fiberglass cloth lightly on the vapor barrier. Make sure you have sealant around the edge of the fiberglass cloth.

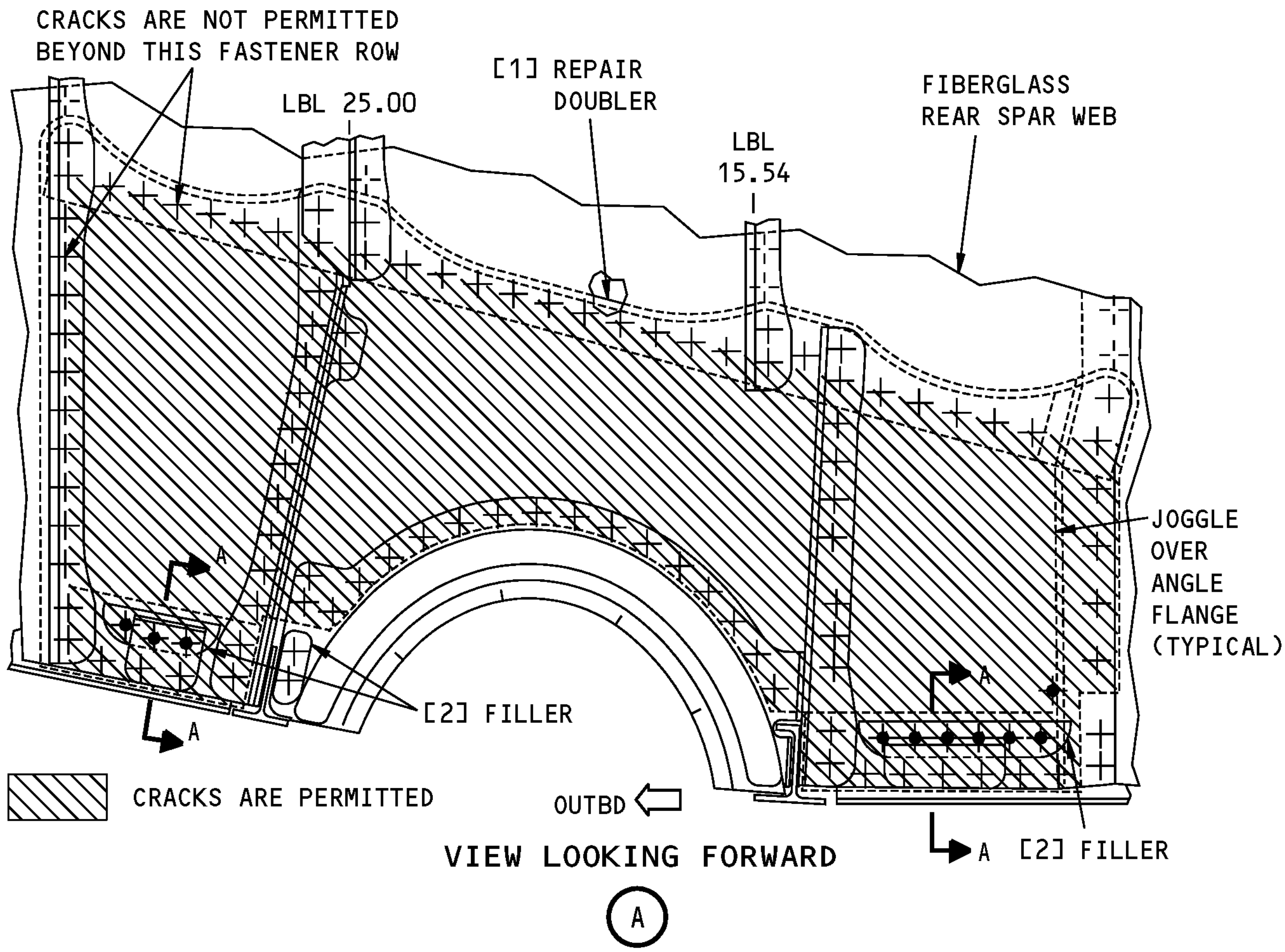
(6) Apply an additional .03/.10 inch (0.76/2.5 mm) thick layer of sealant over the cloth. Make sure the sealant completely covers the fiberglass cloth. Refer to 51-20-05, GENERAL

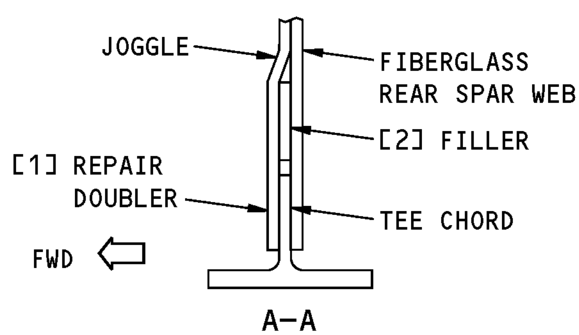
P. Apply a layer of BMS 10-11, Type I primer over the vapor barrier web assembly, all repair parts, and the sealant. Refer to SOPM 20-41-02.

Q. Apply a layer of BMS 5-81 coating as given in the Figure 204 instructions to all vapor barrier web locations after installation and fillet sealing of the repair parts. Refer to AMM SUBJECT 28-11-00.




**Wing Center Section Rear Spar**
**Figure 202**



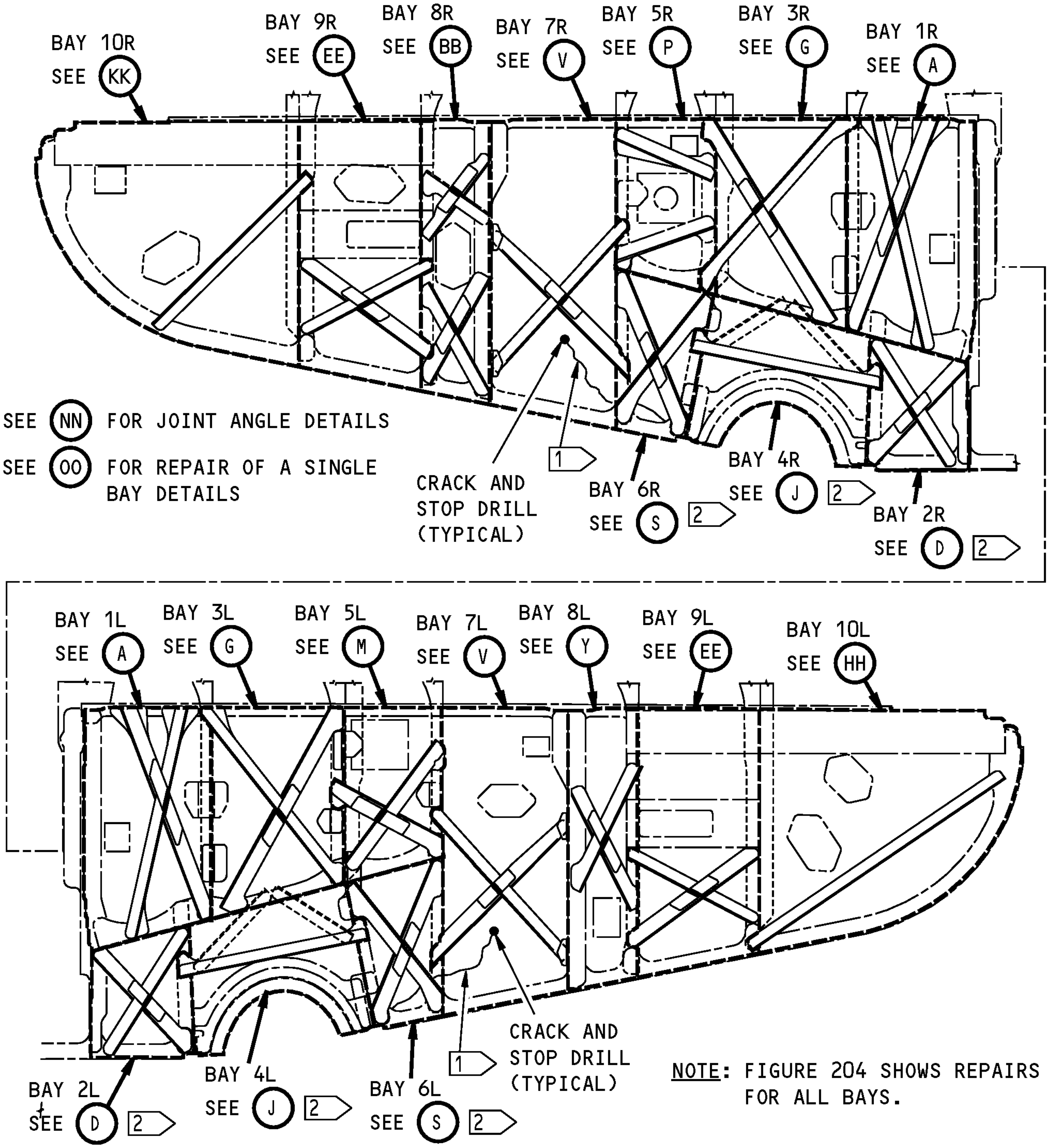




FASTENER SYMBOLS

REPAIR FASTENER LOCATION. INSTALL A BACR15BB5A()C RIVET.

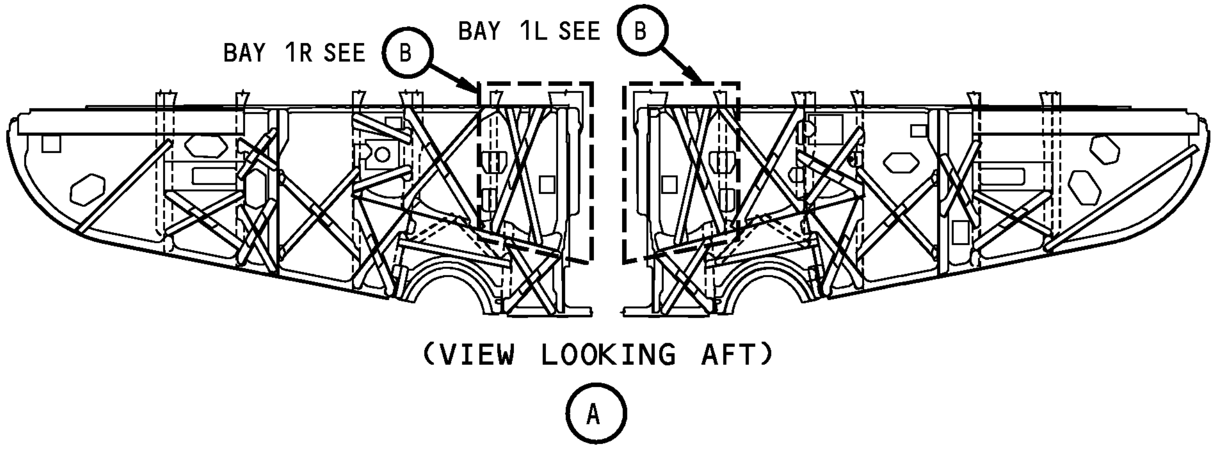
INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AS INTITIAL FASTENER. IT IS PERMITTED TO INSTALL A 1/64 INCH OR 1/32 INCH DIAMETER OVERSIZE FASTENER. MAKE SURE THERE IS A MINIMUM OF 1.5D (D = FASTENER DIAMETER) EDGE MARGIN AND A 4D - 6D DISTANCE BETWEEN FASTENERS.

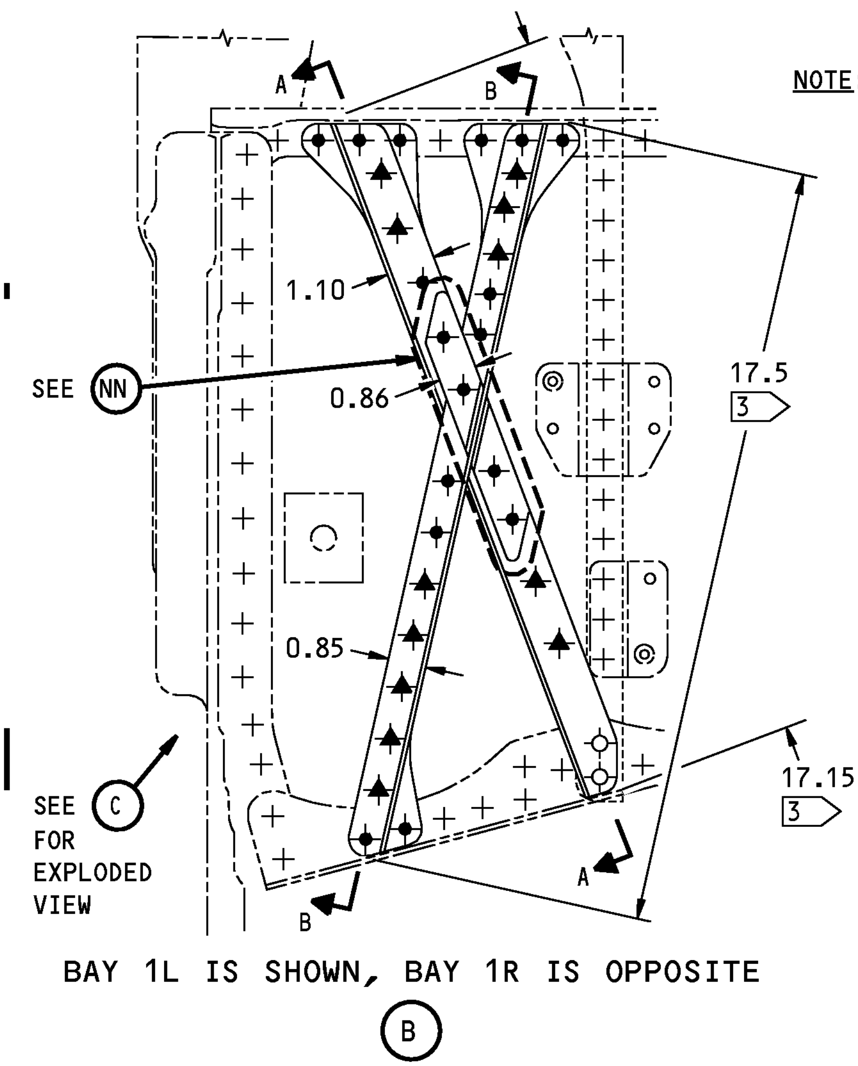
**Wing Center Section Vapor Barrier Repair**
**Figure 203**




CONFIGURATION FOR LINE NUMBERS 1-1639 IS SHOWN
LINE NUMBERS 1640 AND ON ARE SIMILAR
(VIEW LOOKING AFT)

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 1 of 43)**

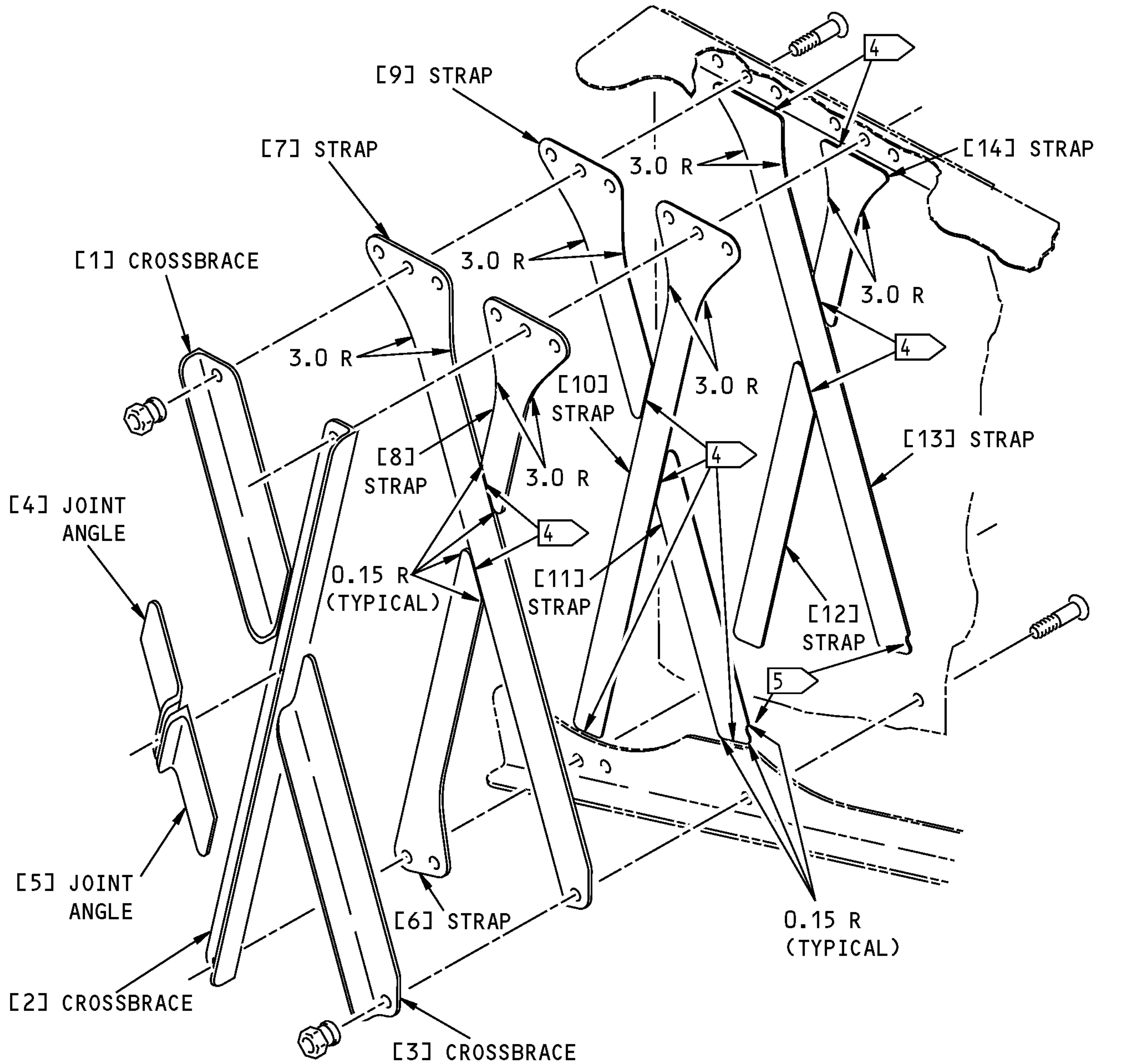



NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85147 S0000178826_V2

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 2 of 43)**

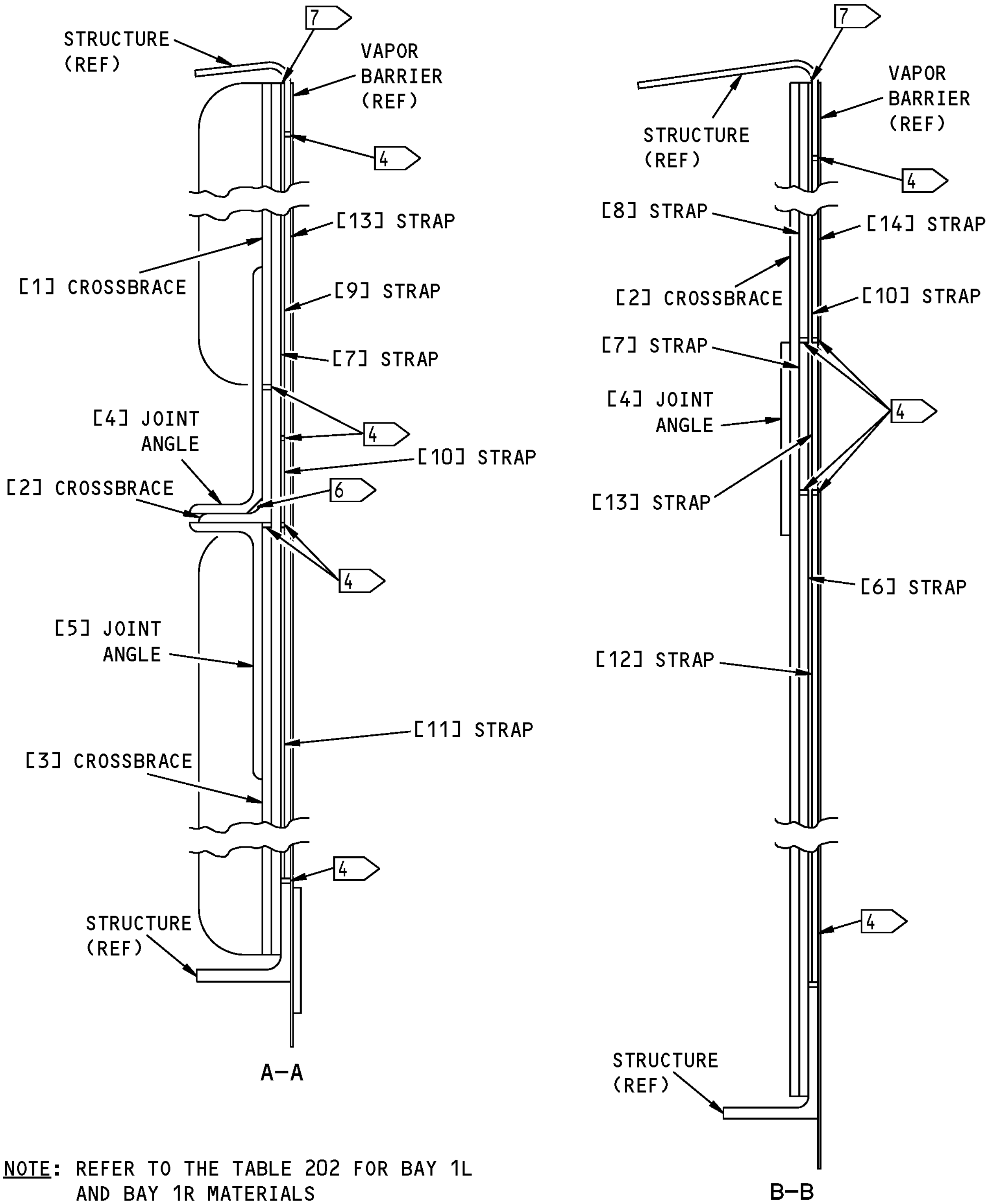



NOTE: REFER TO THE TABLE 202 FOR BAY 1L AND BAY 1R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

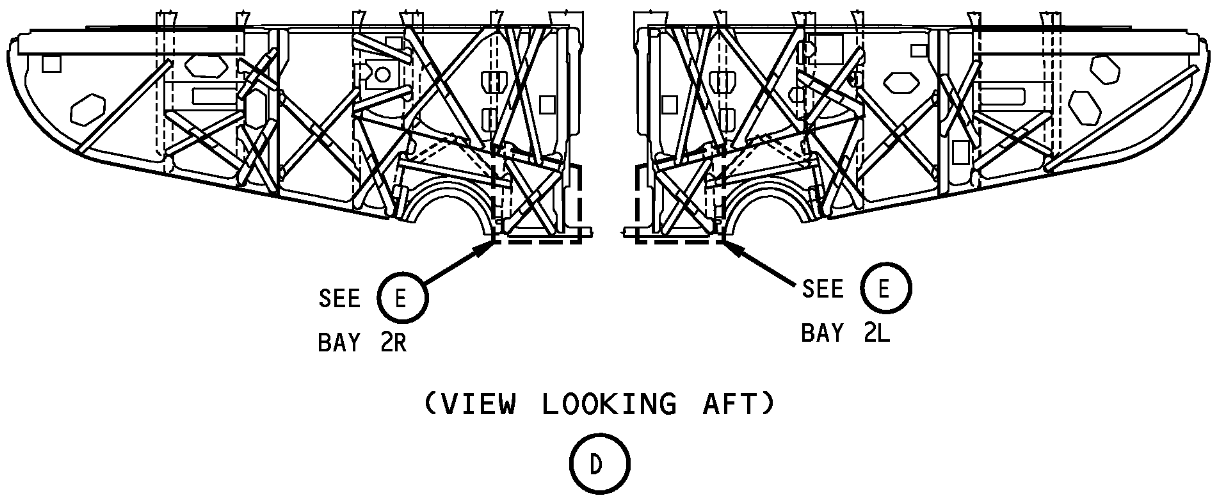
C

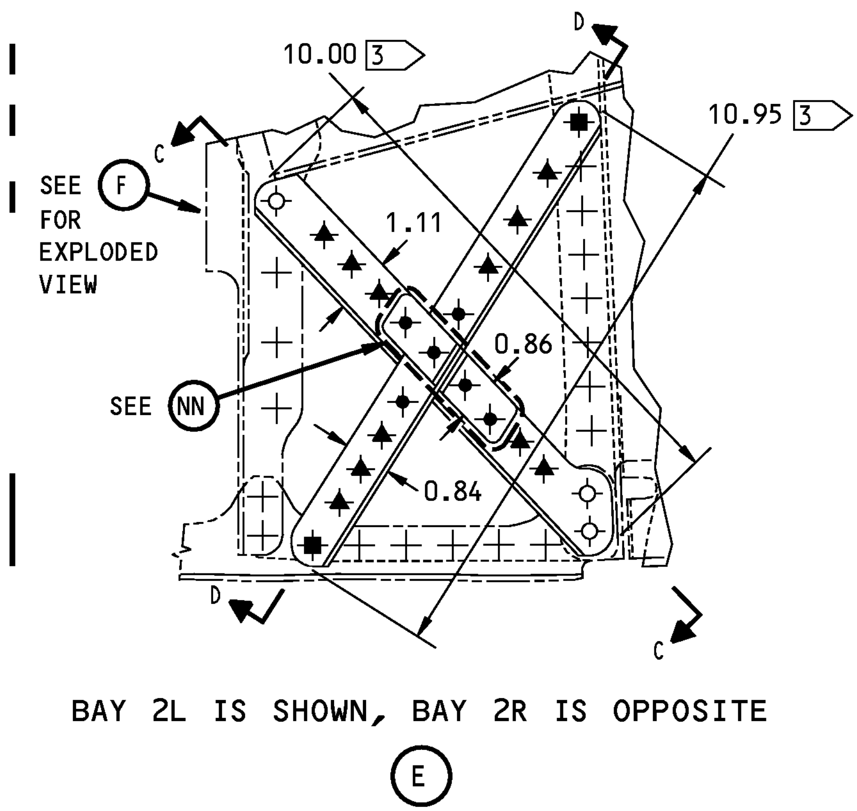
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 3 of 43)**




NOTE: REFER TO THE TABLE 202 FOR BAY 1L AND BAY 1R MATERIALS

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 4 of 43)

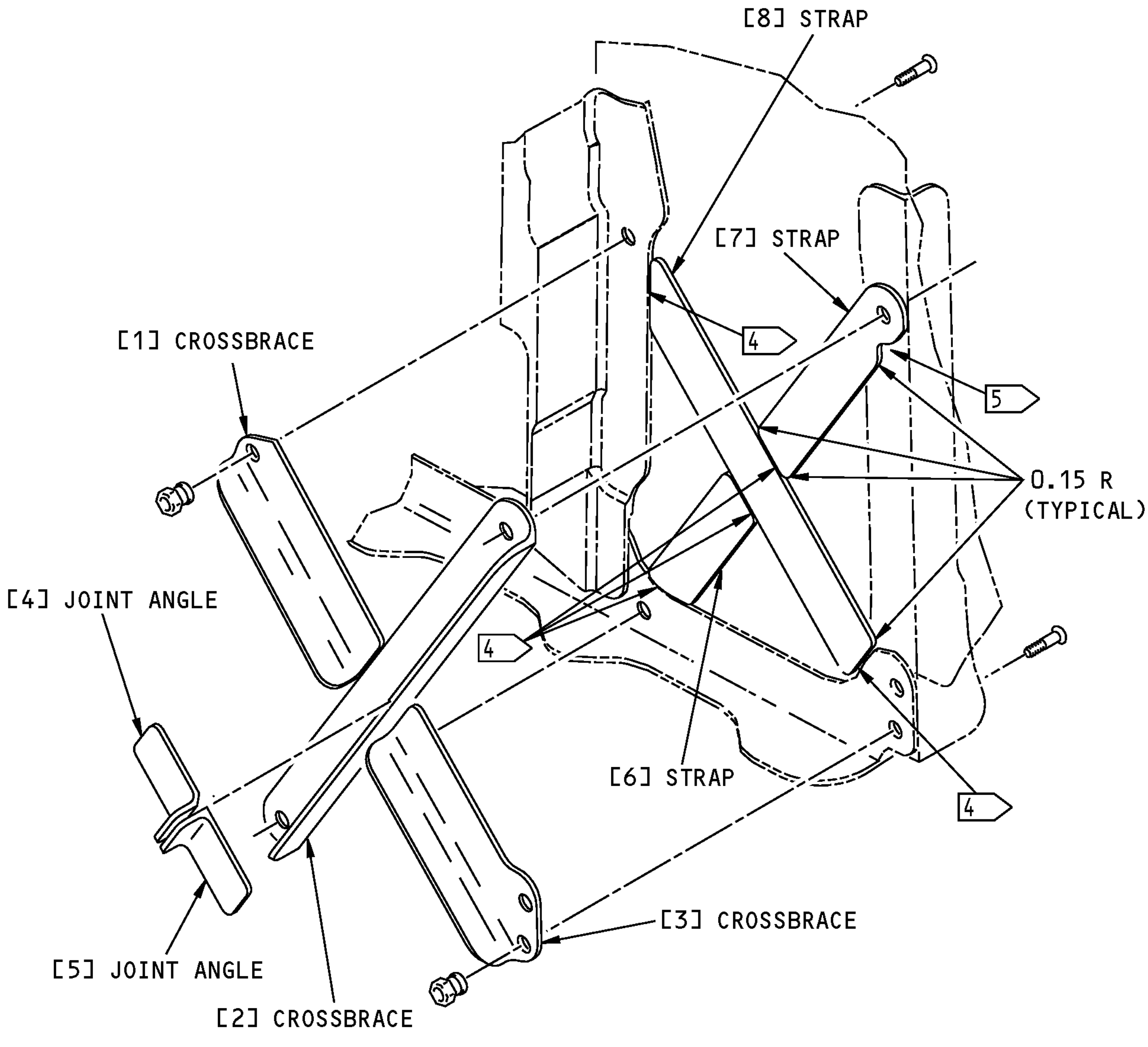



NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85499 S0000181658_V2

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 5 of 43)**

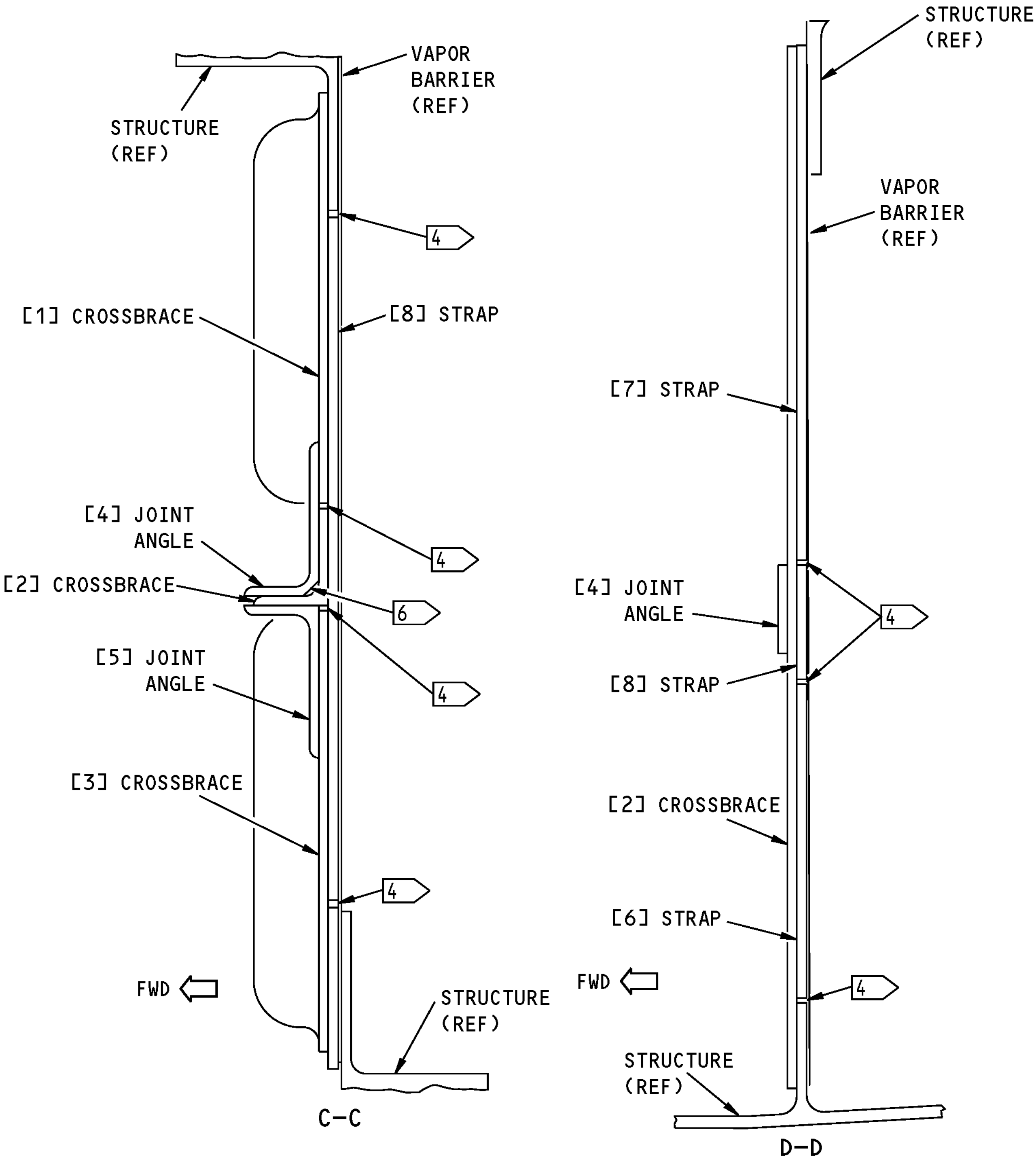



NOTE: REFER TO THE TABLE 203 FOR BAY 2L AND BAY 2R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

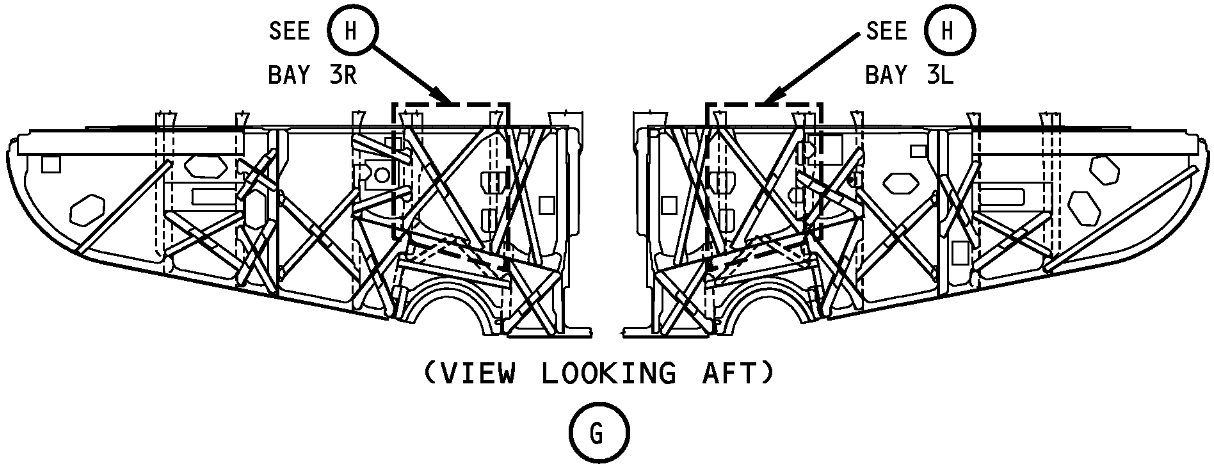
F

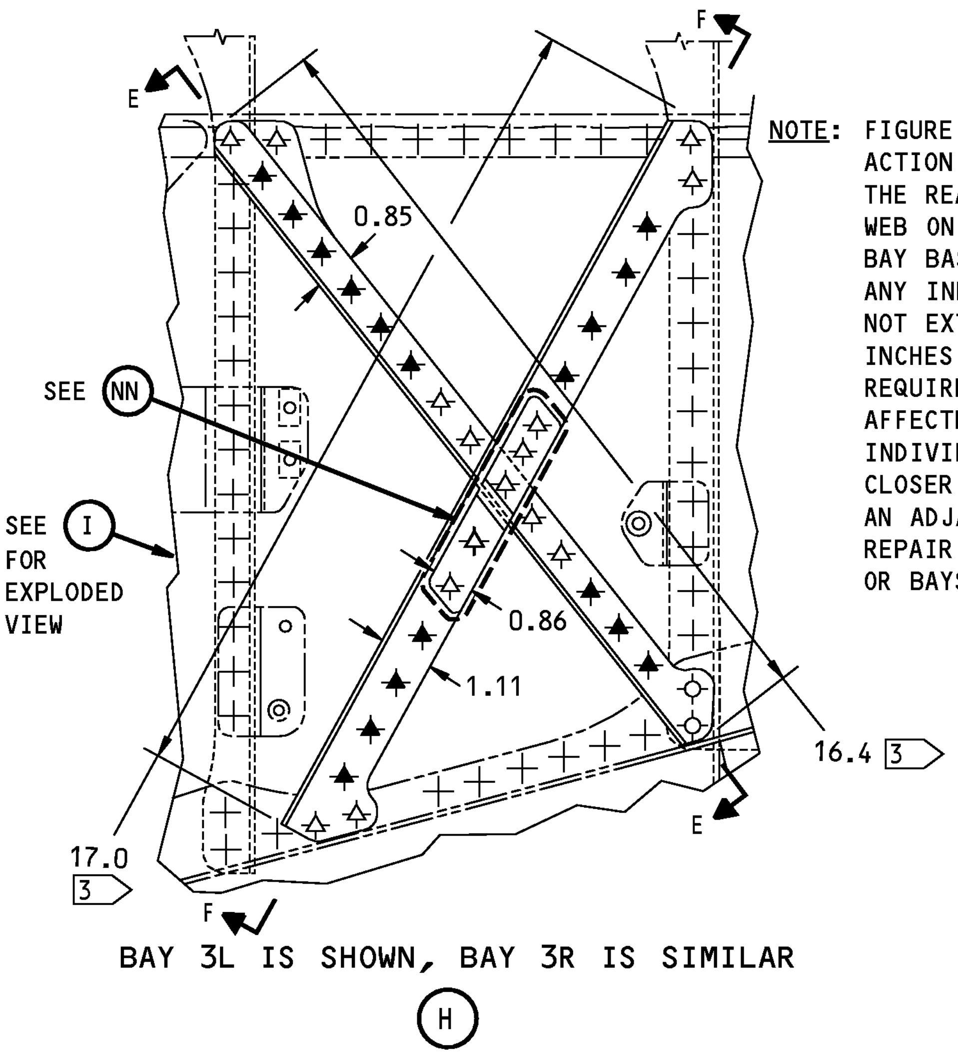
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 6 of 43)**




NOTE: REFER TO THE TABLE 203 FOR BAY 2L
AND BAY 2R MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 7 of 43)**

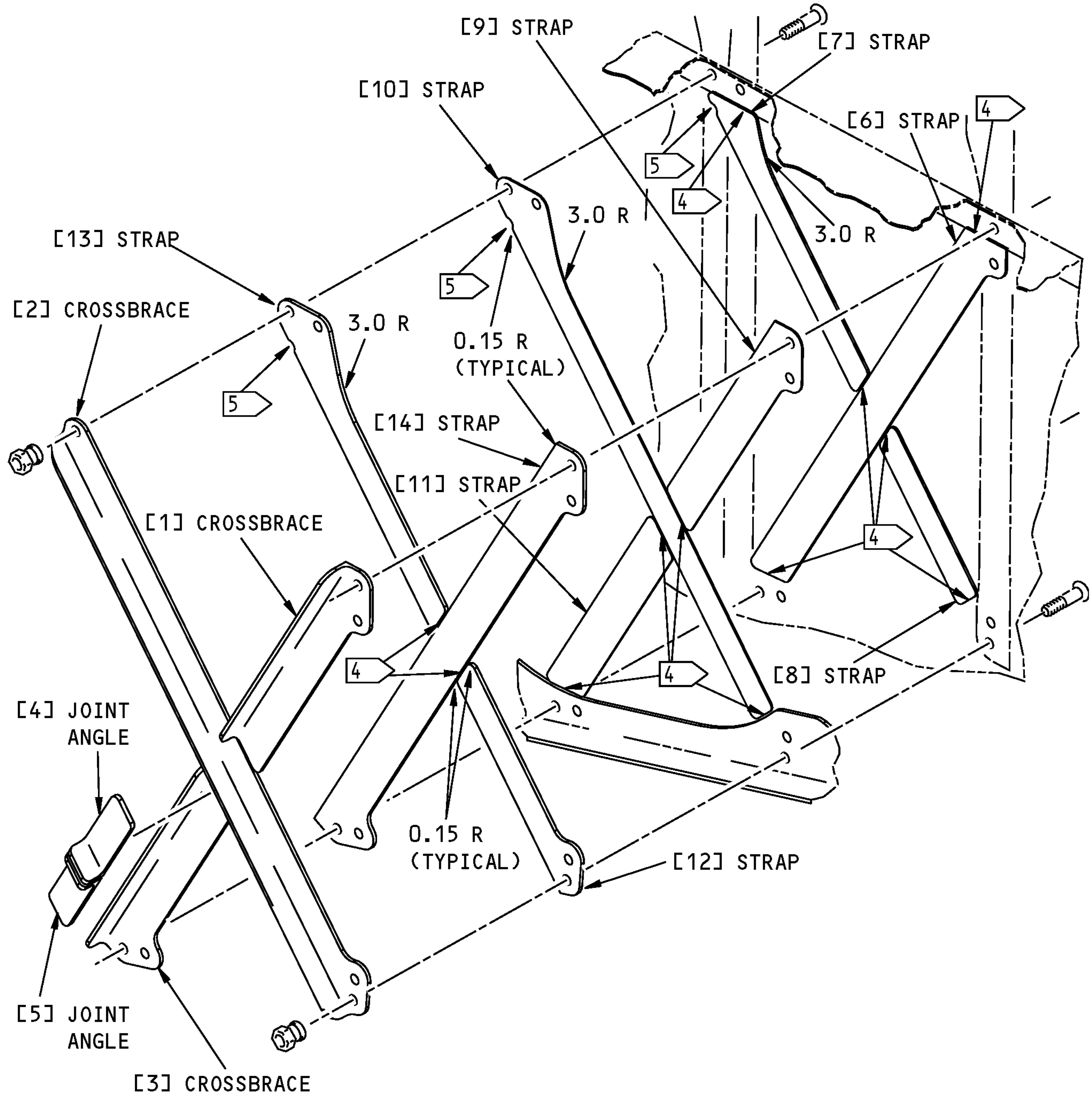



NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85502 S0000181659_V2

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 8 of 43)**

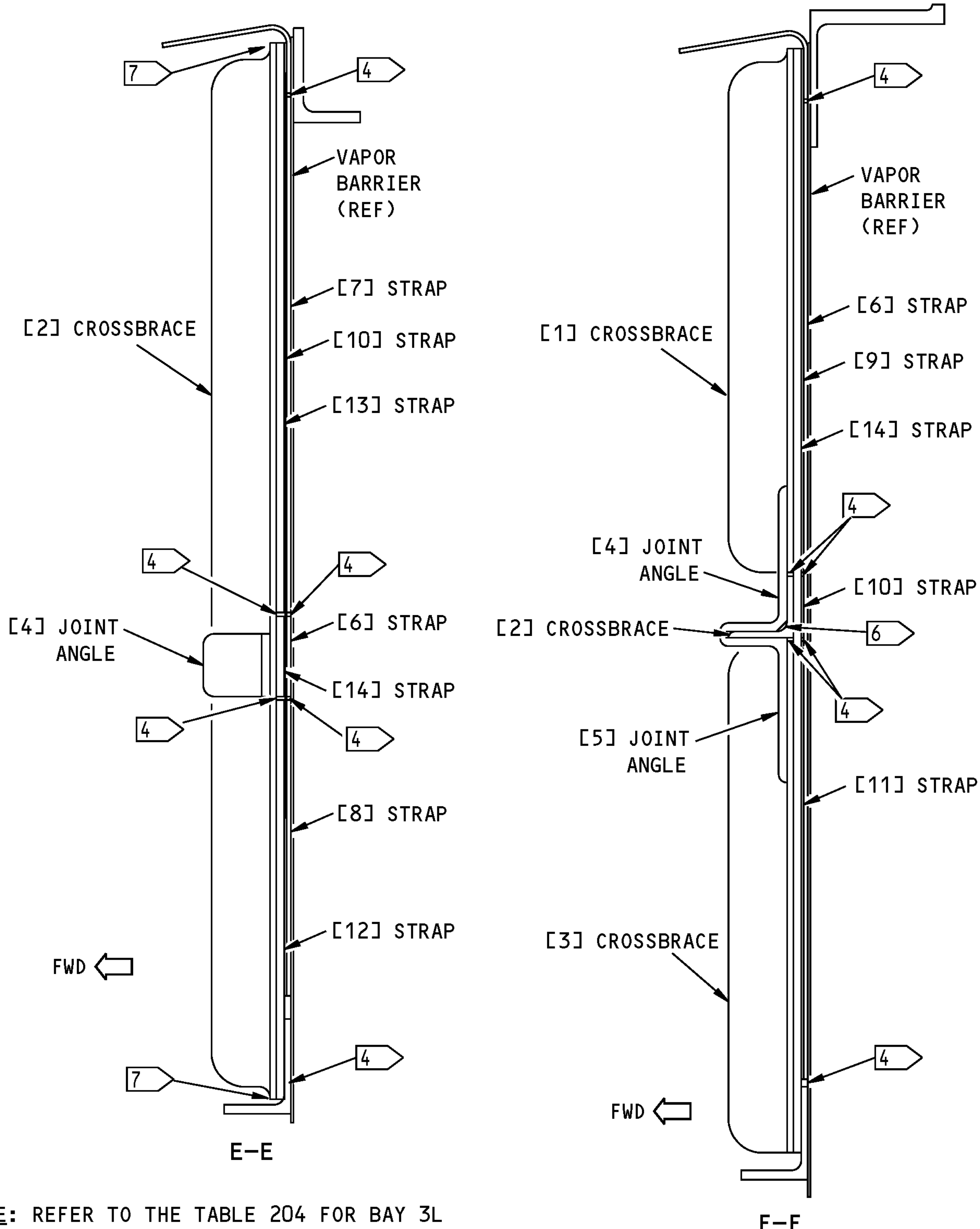



NOTE: REFER TO THE TABLE 204 FOR BAY 3L AND BAY 3R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

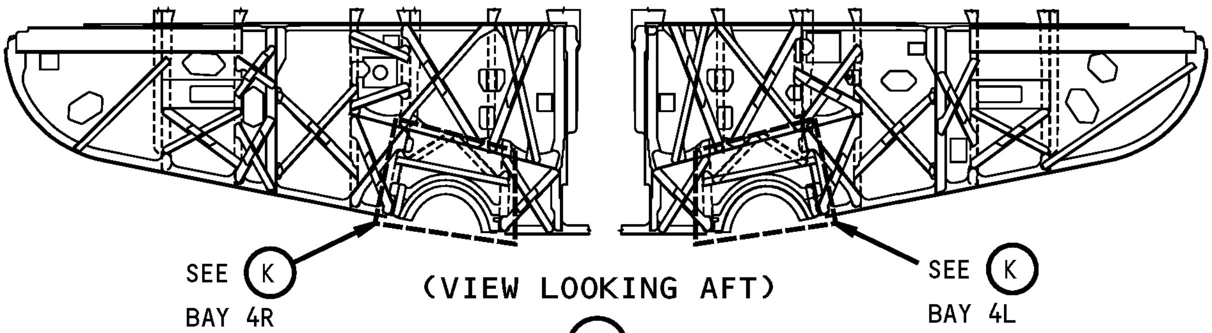
I

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 9 of 43)**

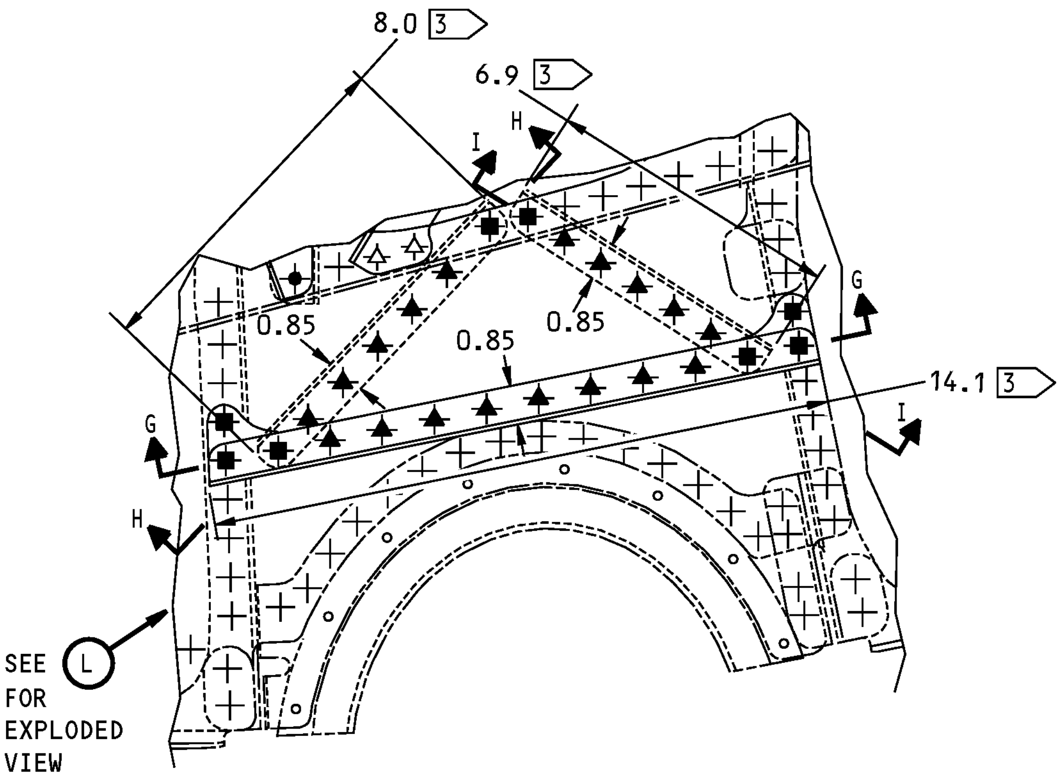



NOTE: REFER TO THE TABLE 204 FOR BAY 3L AND BAY 3R MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 10 of 43)**




J

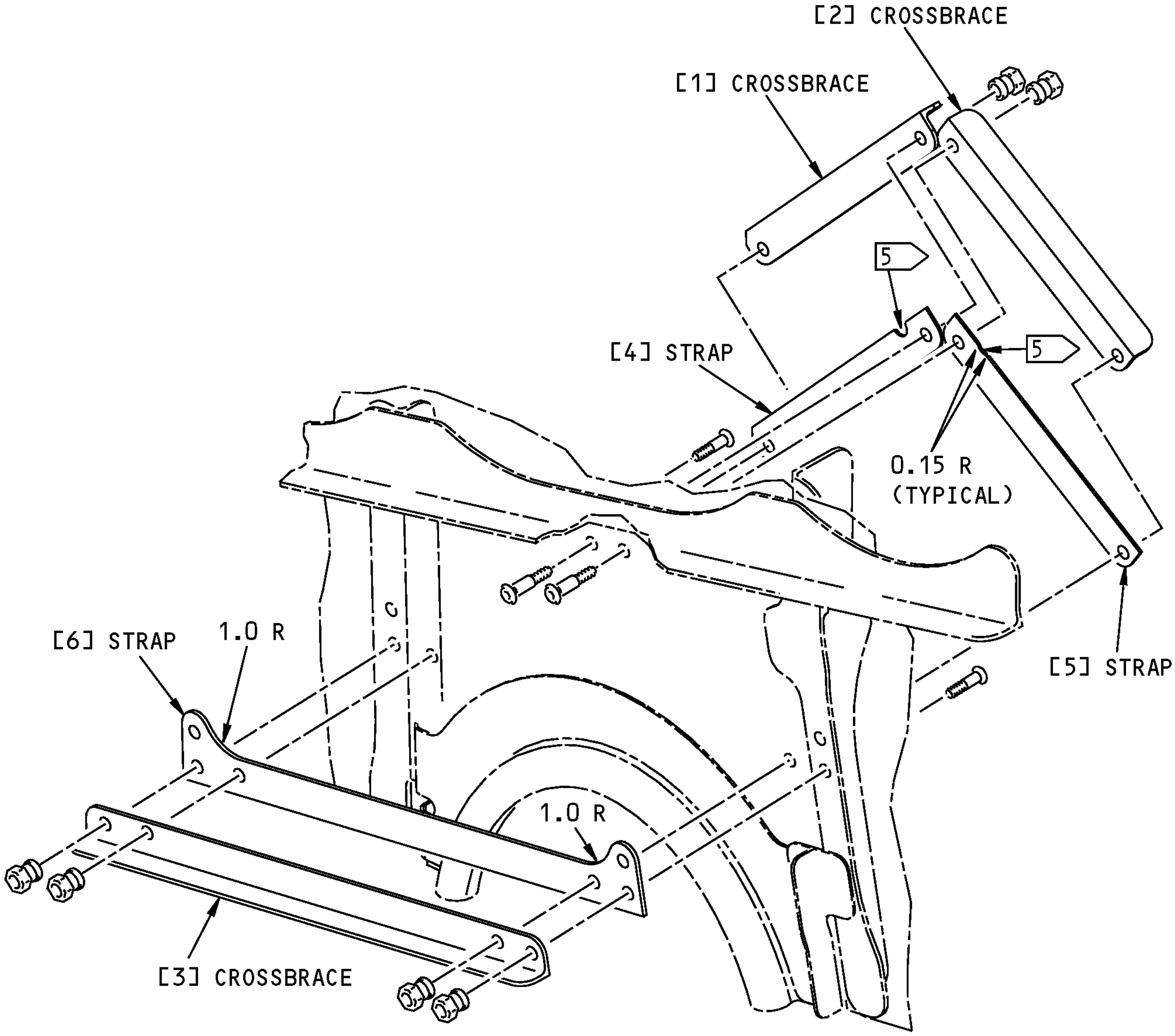



BAY 4L IS SHOWN, BAY 4R IS OPPOSITE

K

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 11 of 43)**

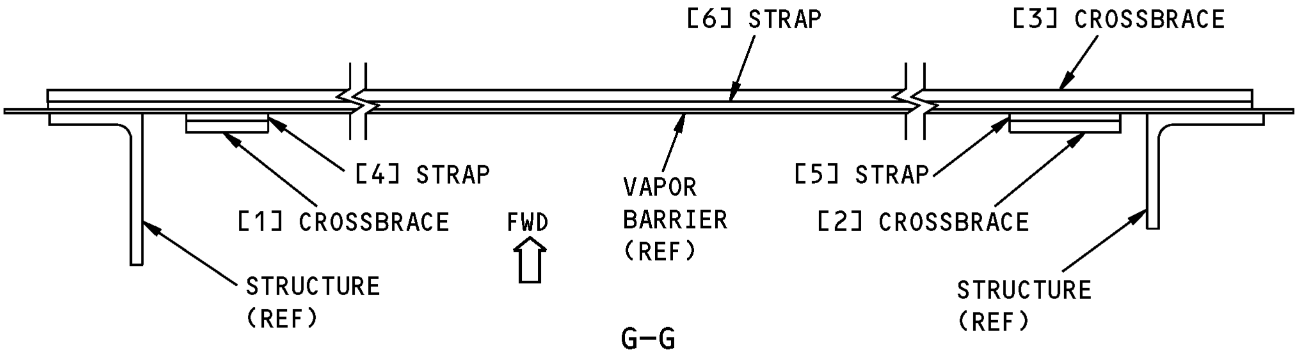


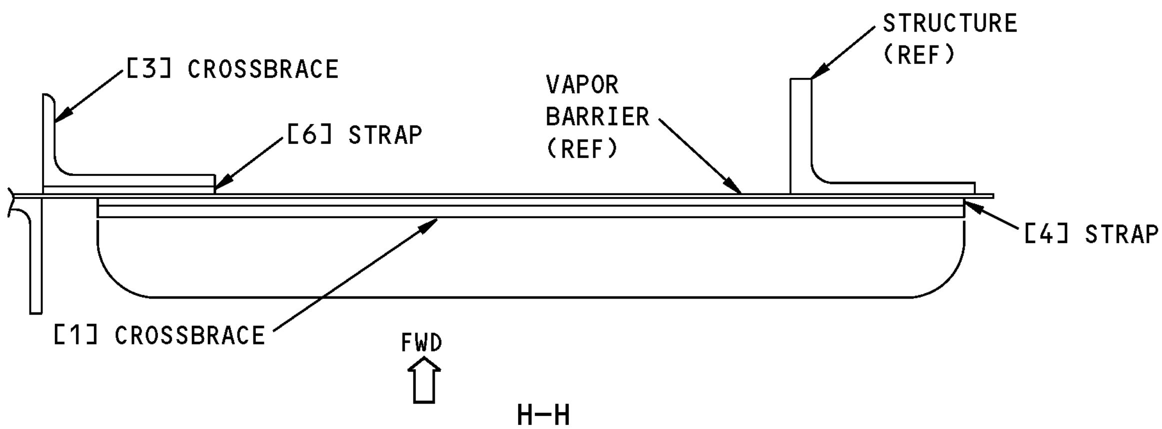

NOTE: REFER TO THE TABLE 205 FOR BAY 4L AND BAY 4R MATERIALS

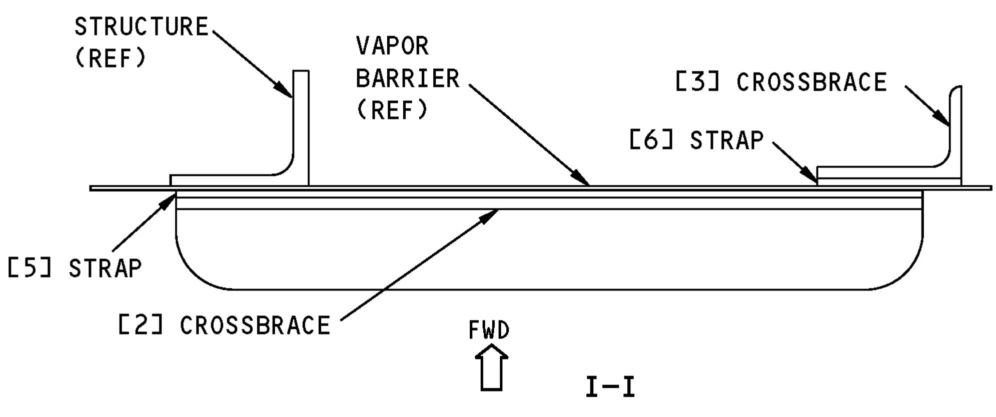
EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

L

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 12 of 43)**

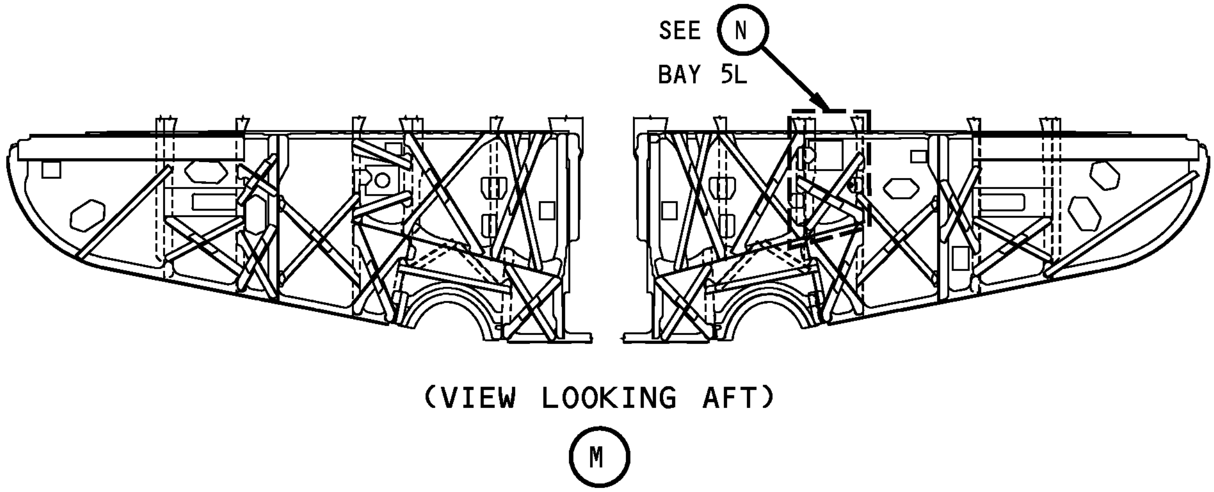


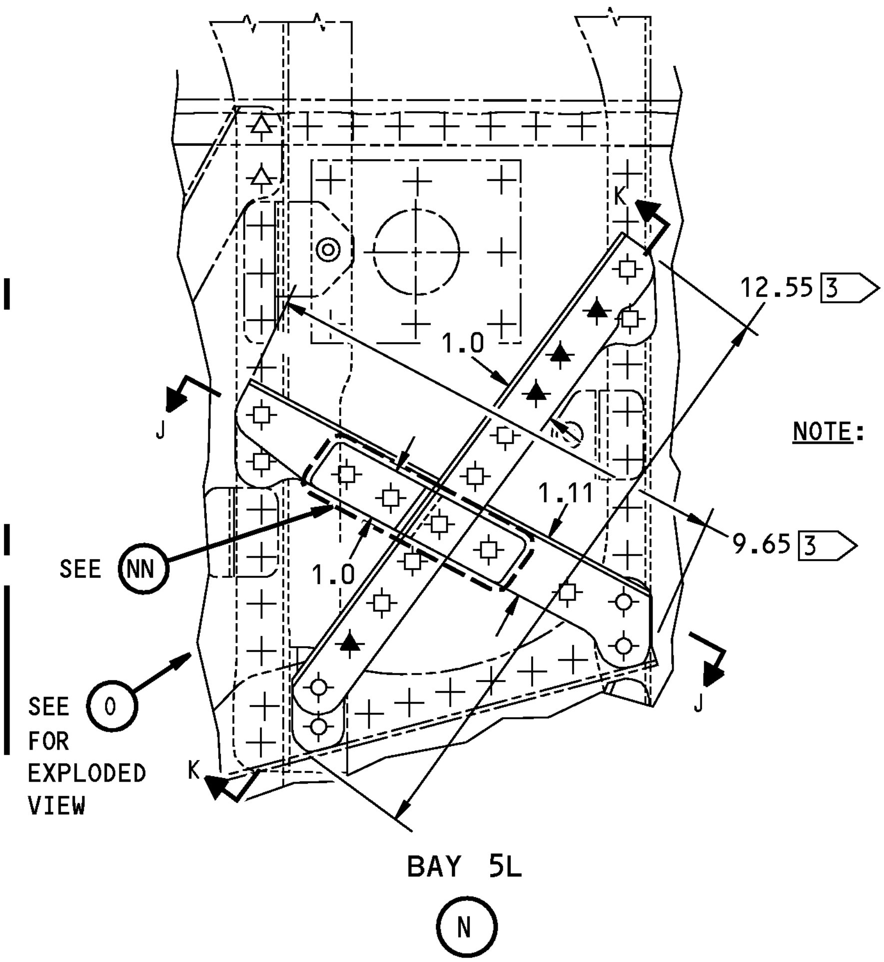




NOTE: REFER TO THE TABLE 205 FOR BAY 4L AND BAY 4R MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 13 of 43)**

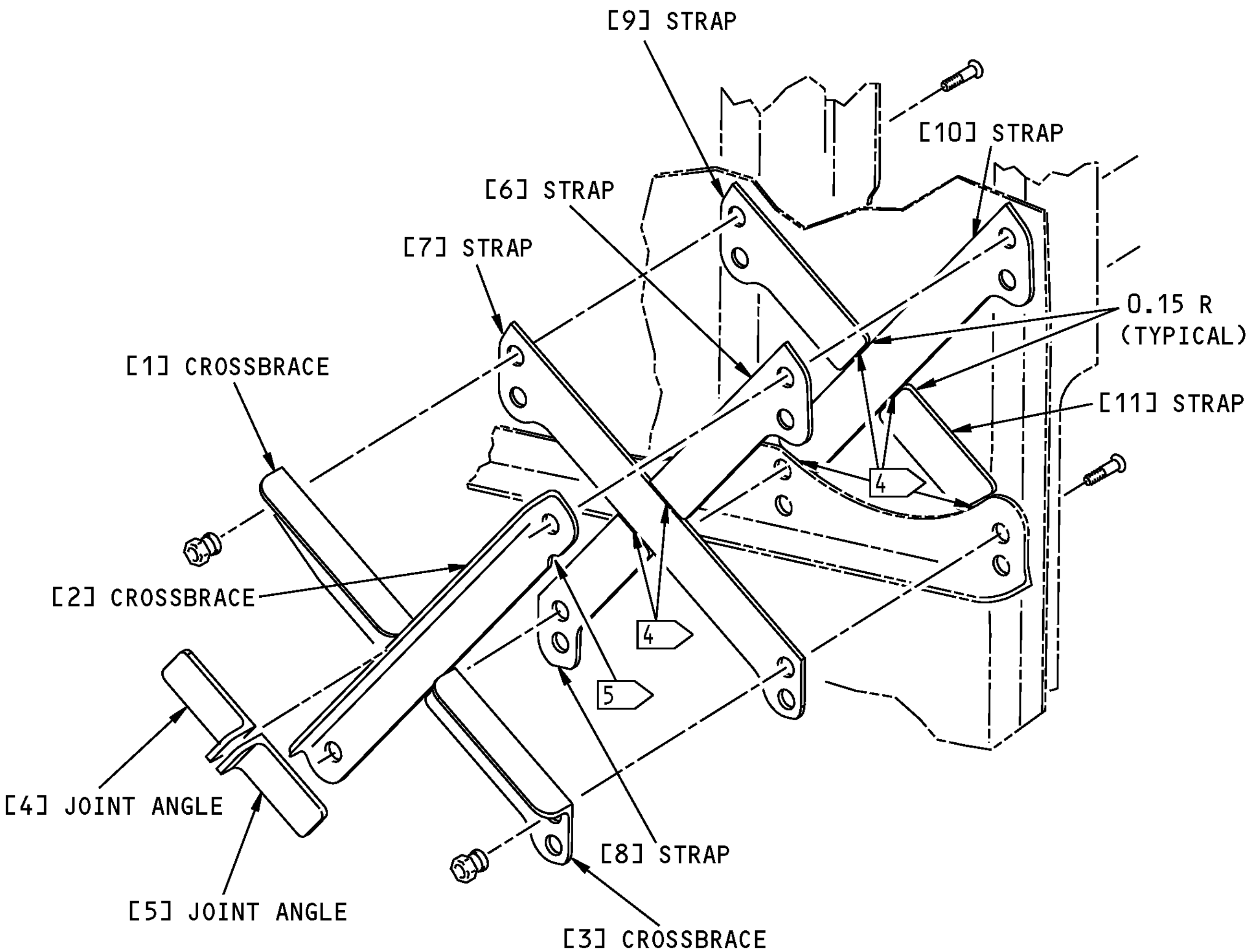



NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85541 S0000181661_V2

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 14 of 43)**

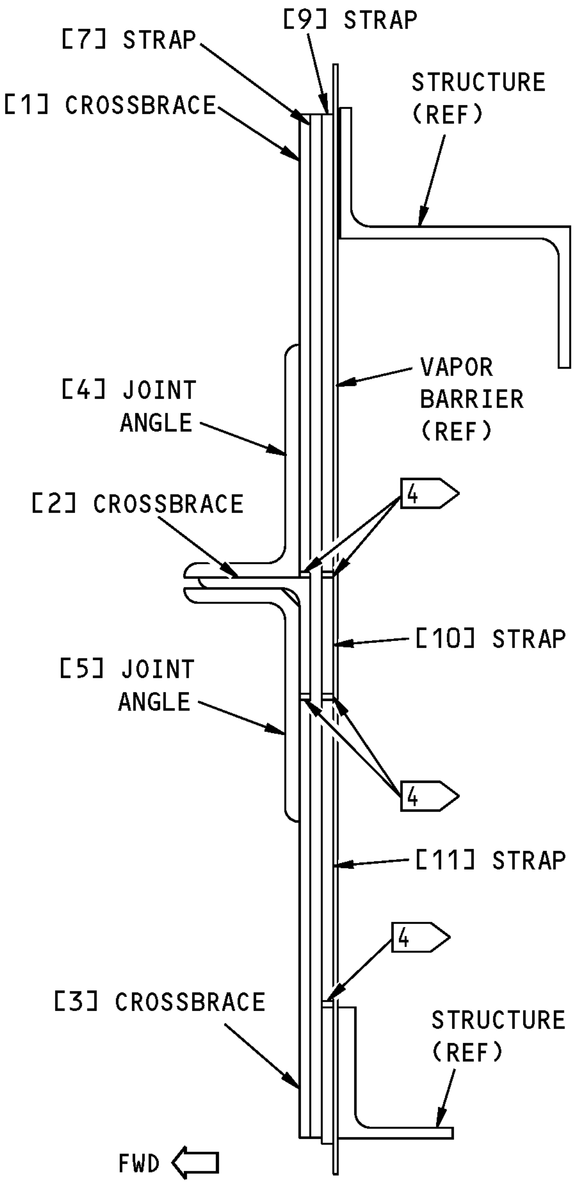



NOTE: REFER TO THE TABLE 206 FOR BAY 5L MATERIALS

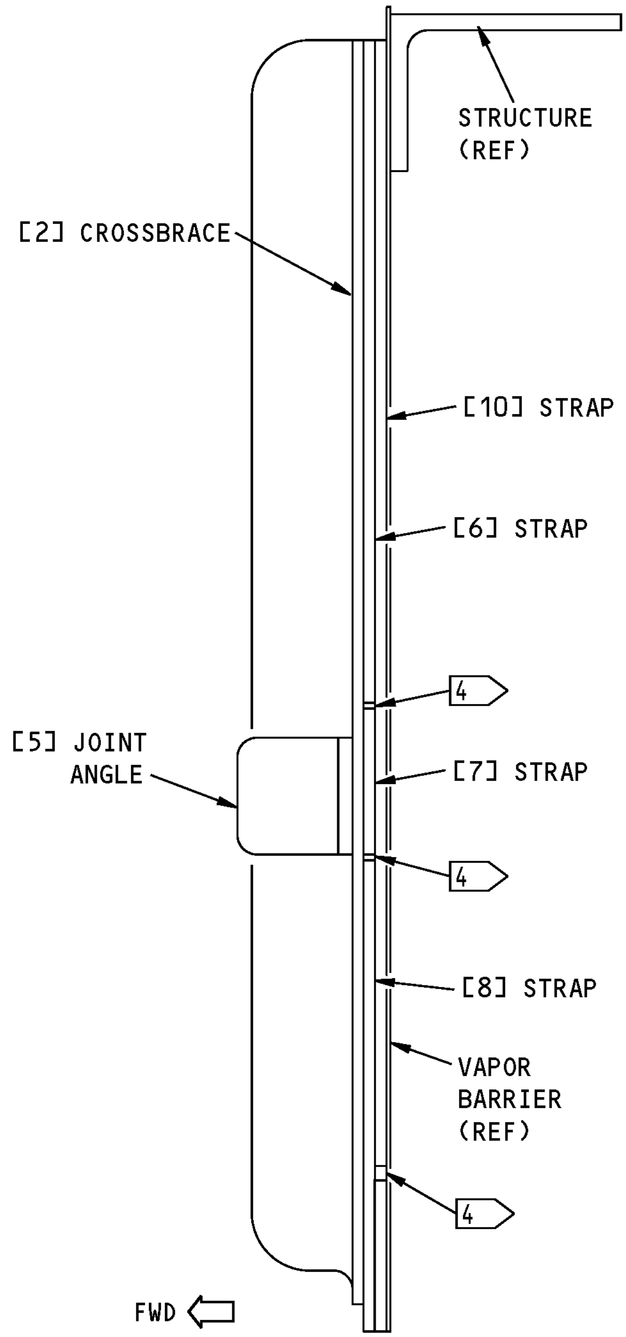
EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

0

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 15 of 43)**



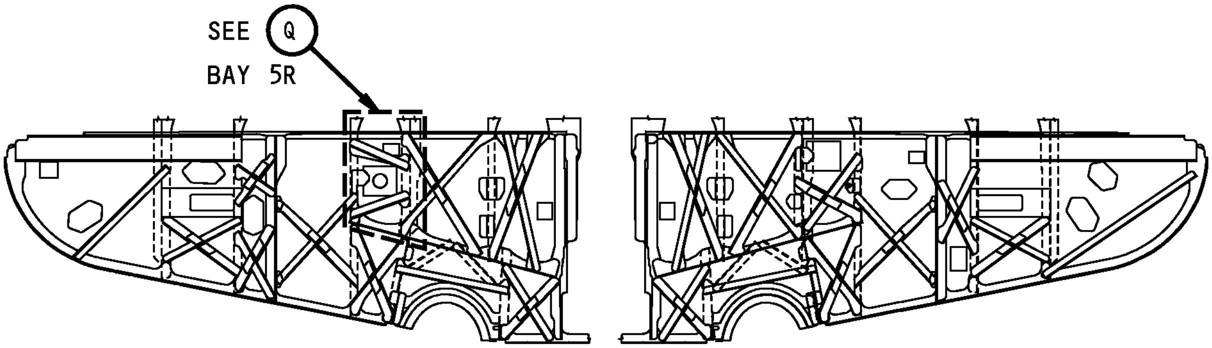

J-J




K-K

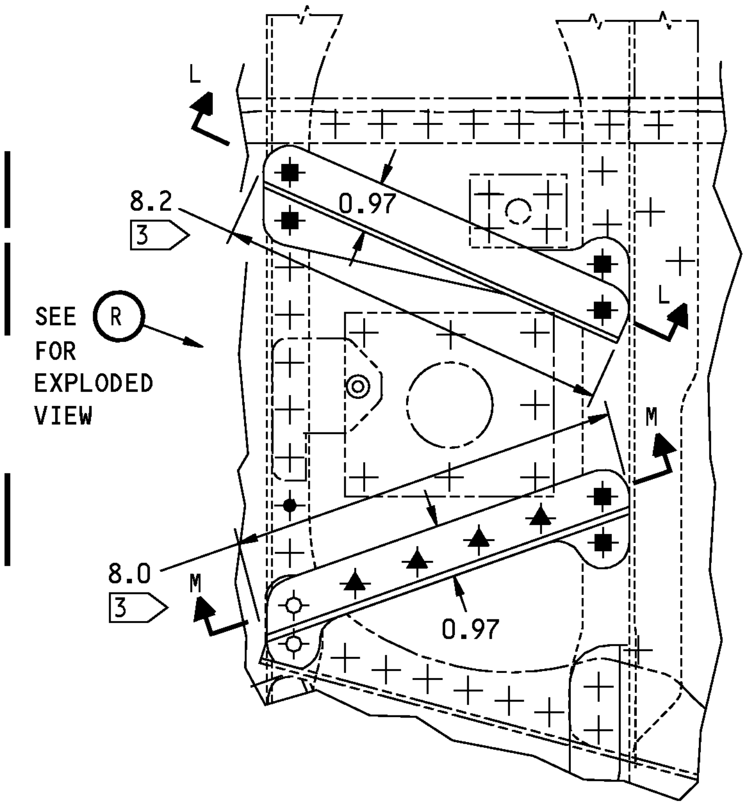
NOTE: REFER TO THE TABLE 206 FOR BAY 5L MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 16 of 43)**




(VIEW LOOKING AFT)

P



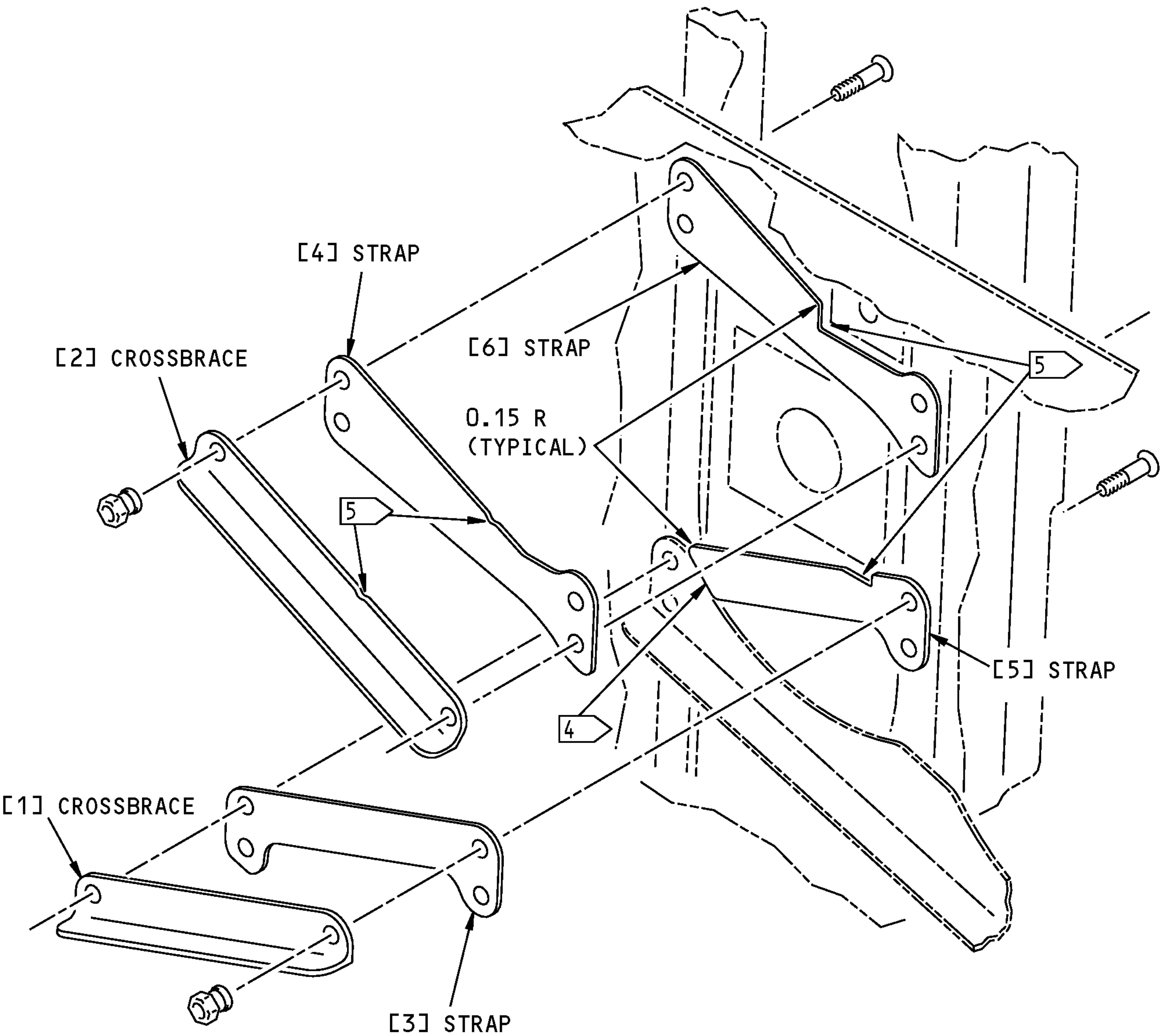

BAY 5R

Q

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85554 S0000181662_V2

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 17 of 43)**

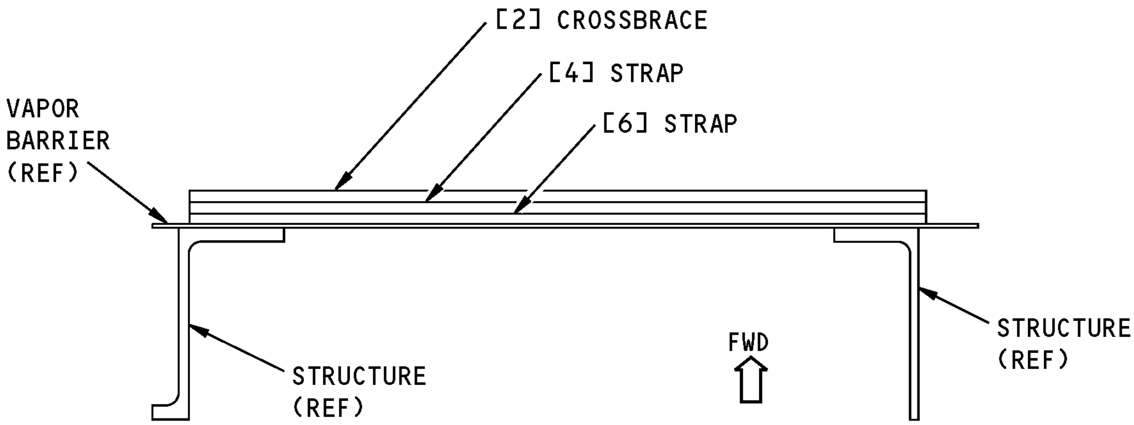



NOTE: REFER TO THE TABLE 207 FOR BAY 5R MATERIALS

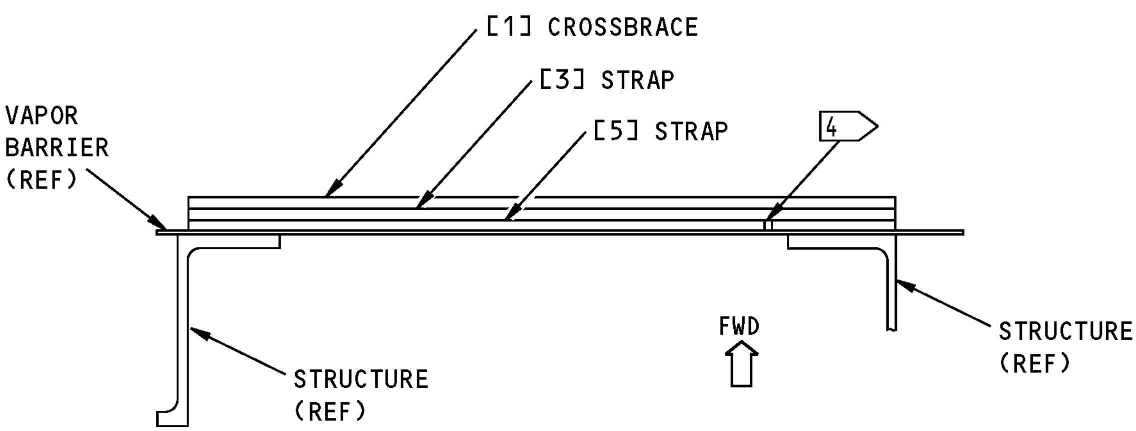
EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

R

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 18 of 43)**



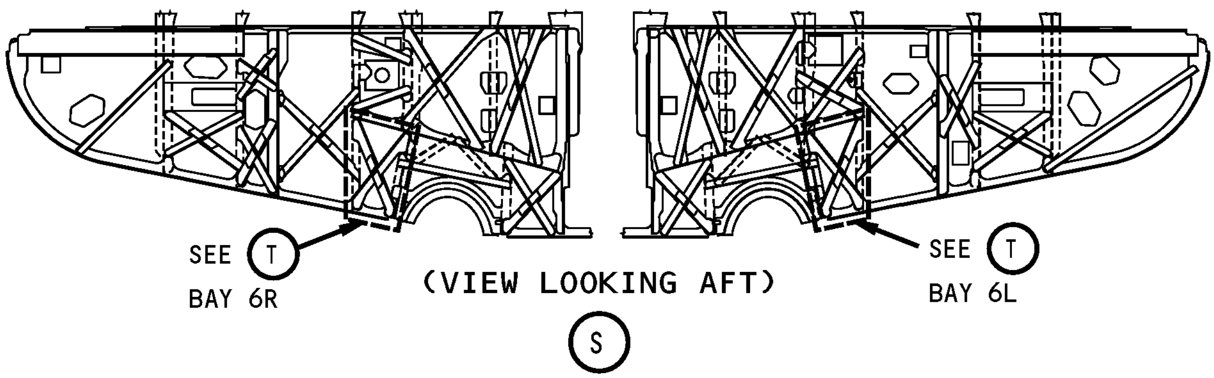

L-L



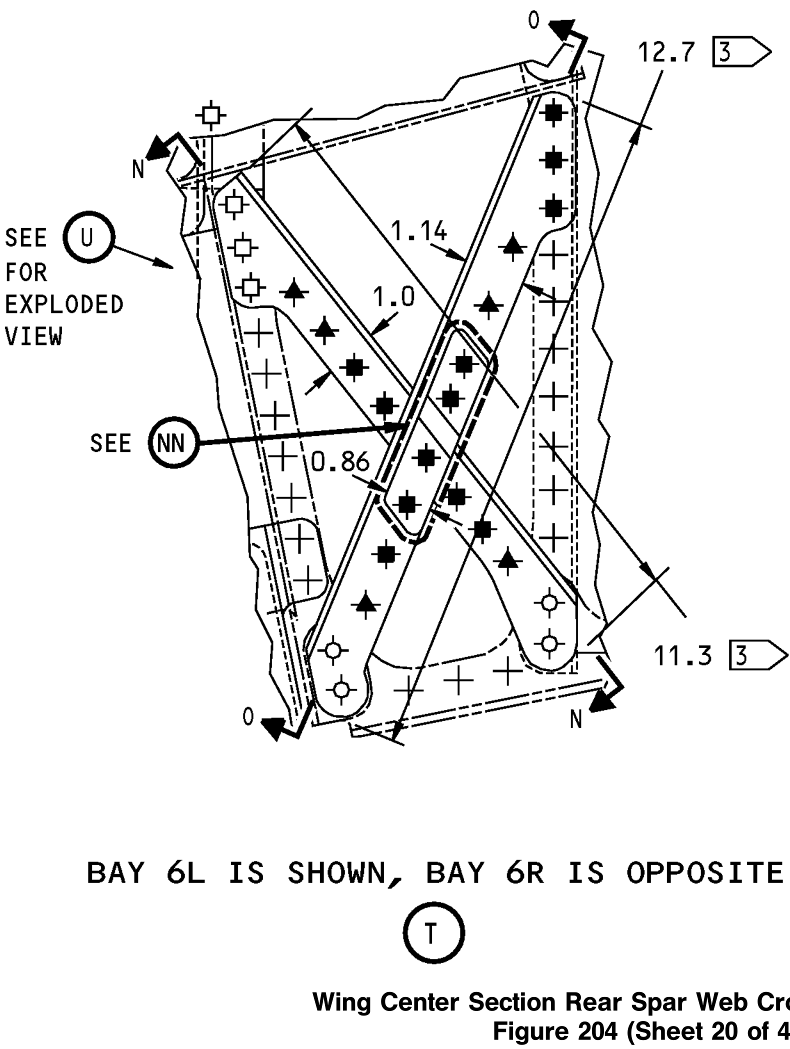

M-M

NOTE: REFER TO THE TABLE 207 FOR BAY 5R MATERIALS

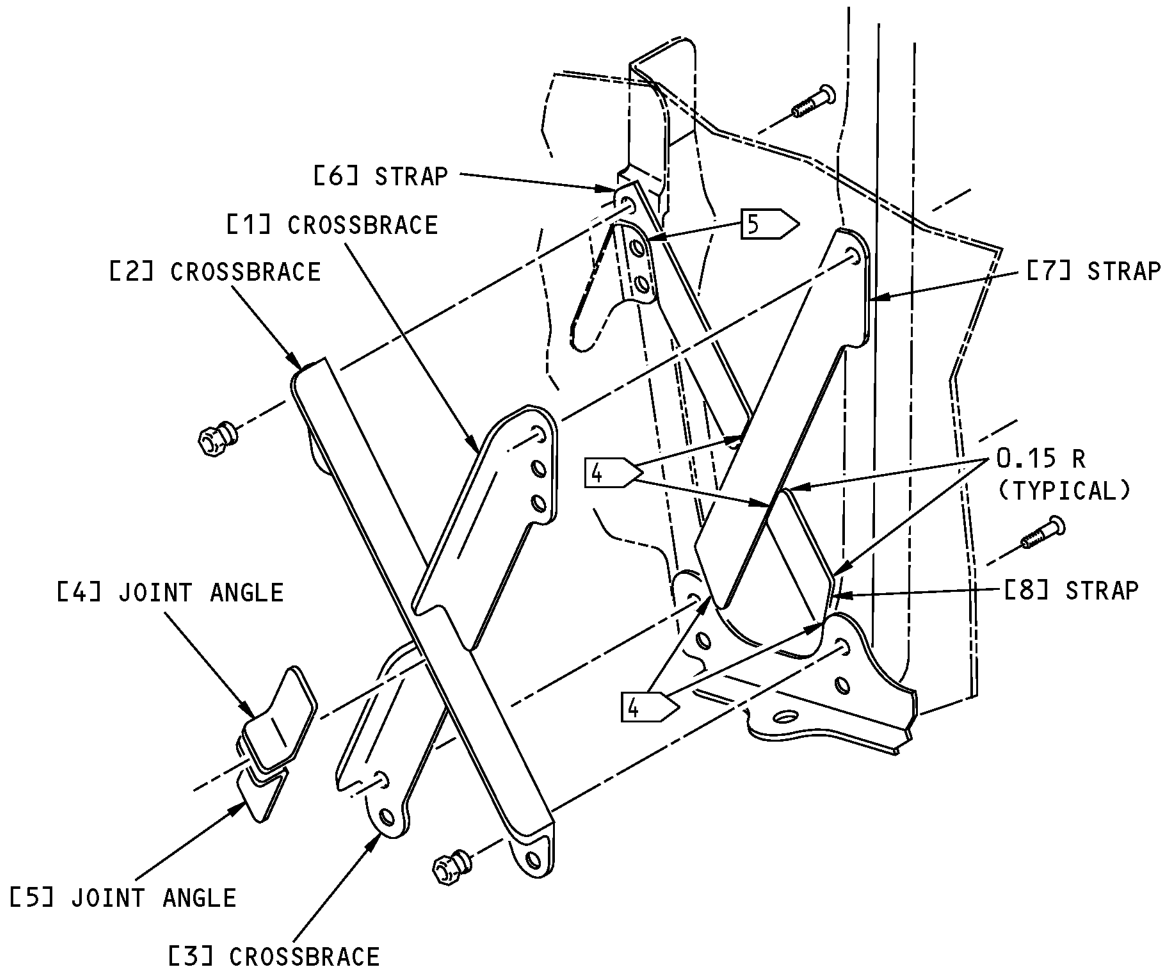
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 19 of 43)**




NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.




**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 20 of 43)**

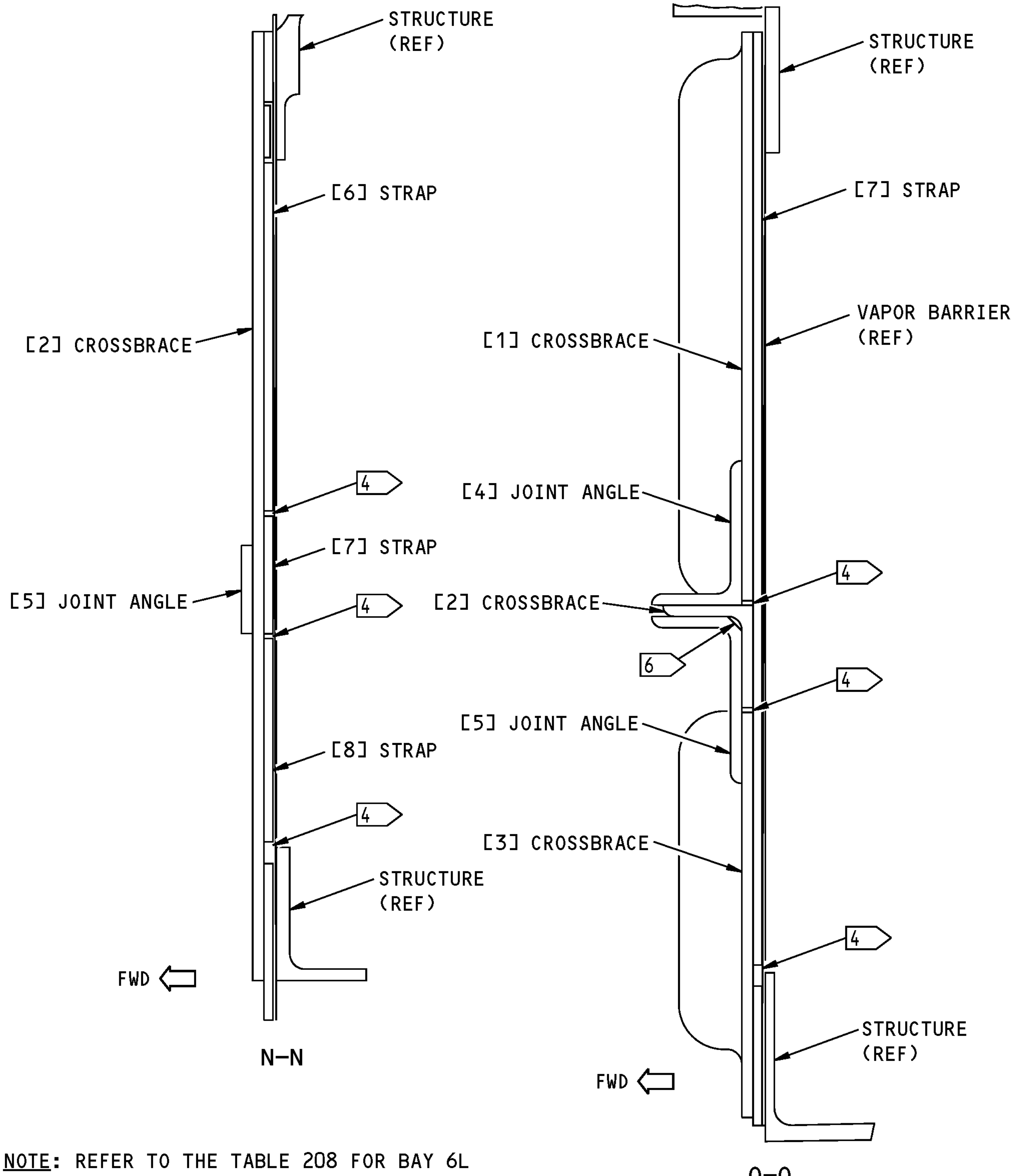



NOTE: REFER TO THE TABLE 208 FOR BAY 6L AND BAY 6R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

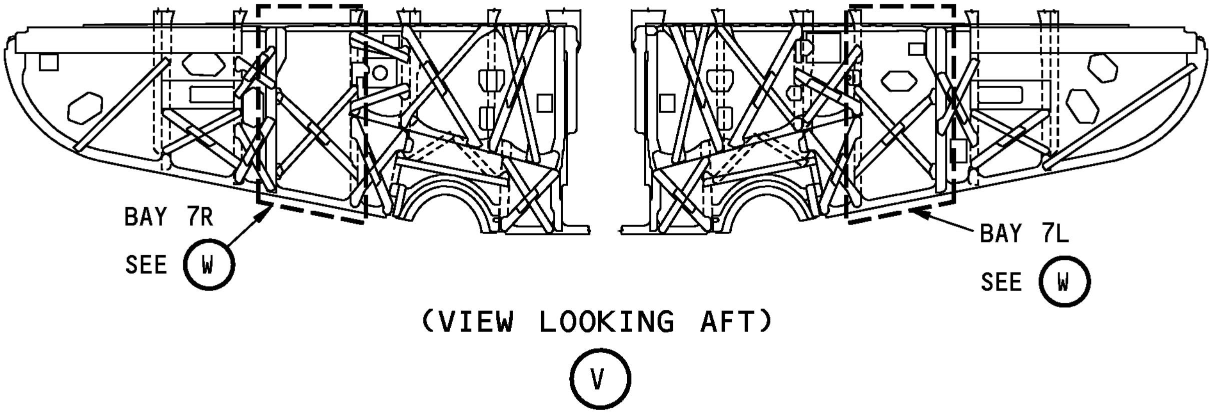
U

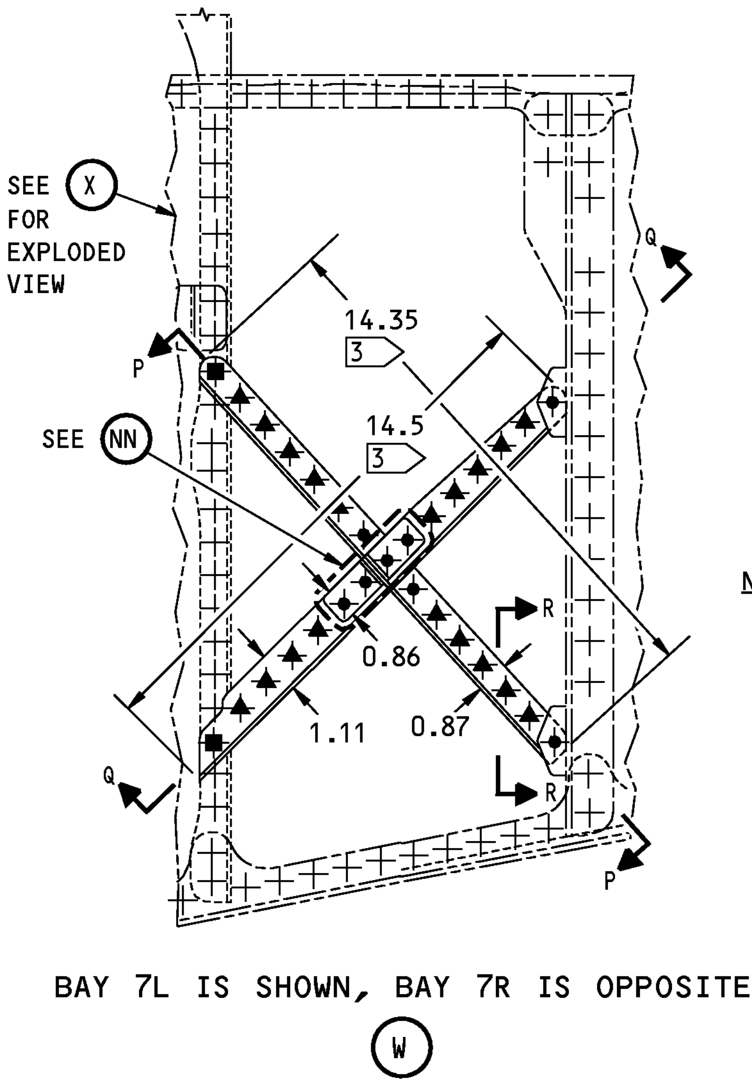
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 21 of 43)**

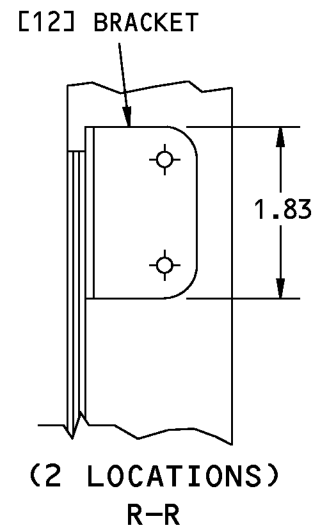



NOTE: REFER TO THE TABLE 208 FOR BAY 6L AND BAY 6R MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 22 of 43)**



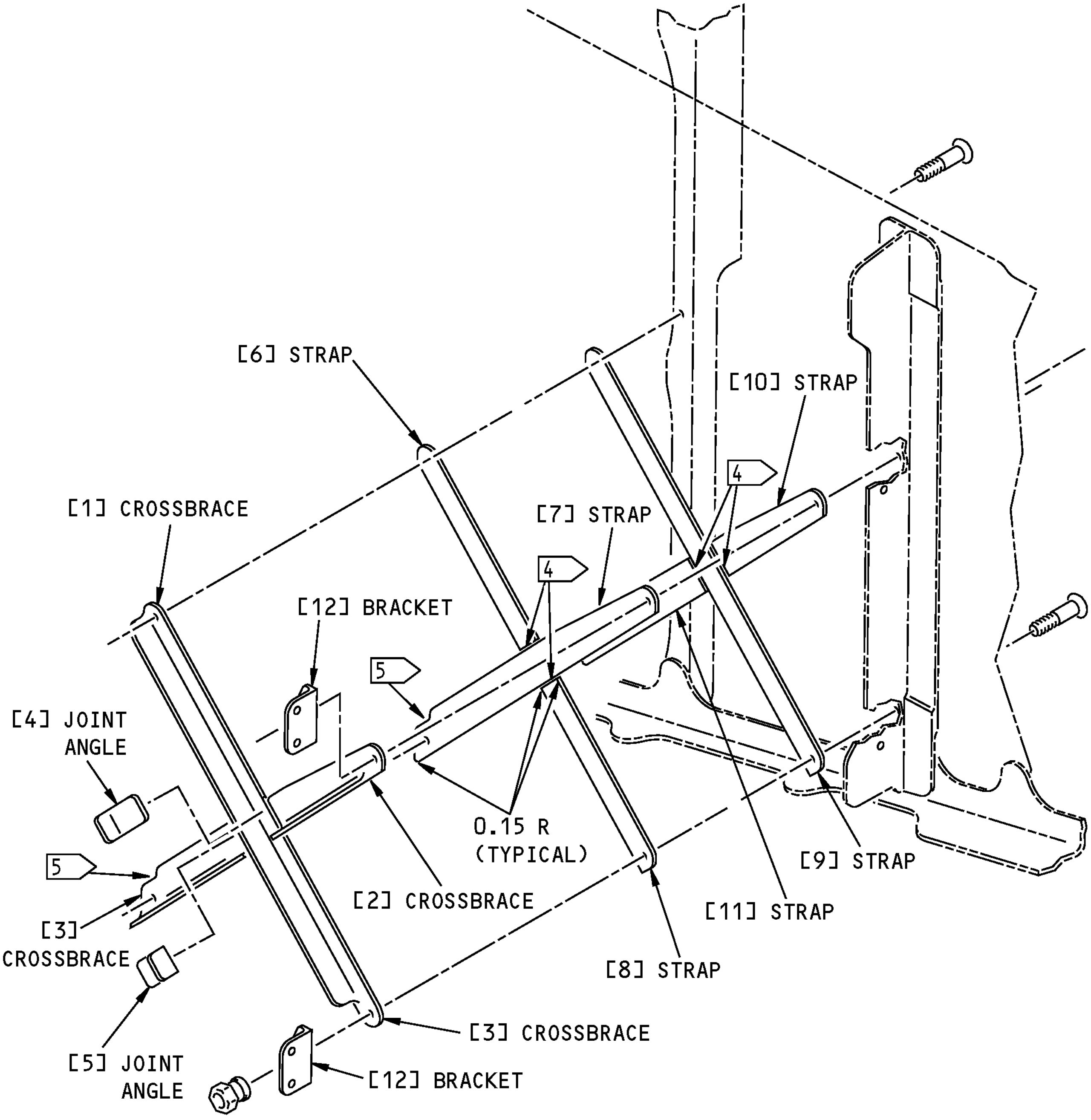




NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85594 S0000181664_V4

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 23 of 43)**

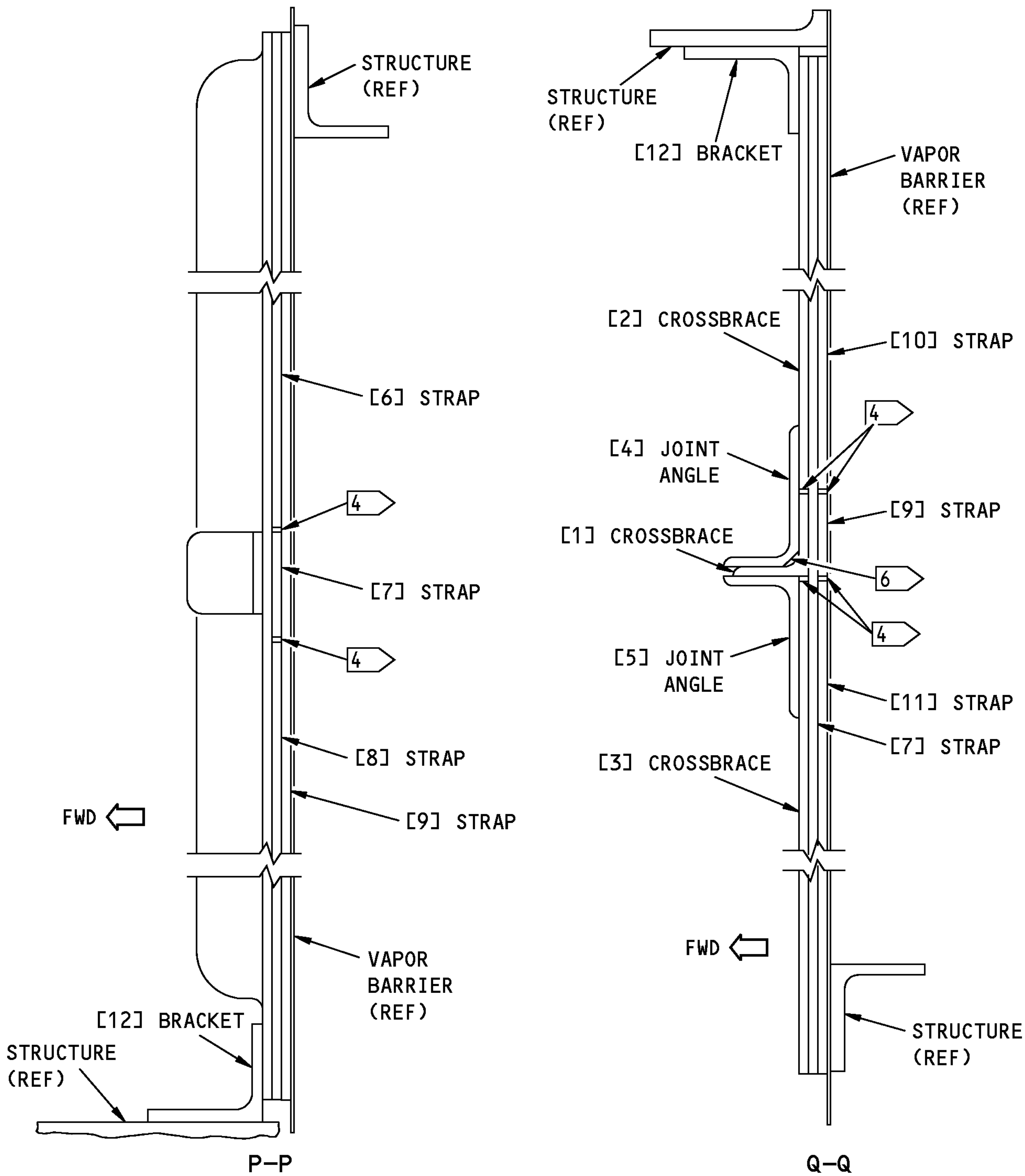



NOTE: REFER TO THE TABLE 209 FOR BAY 7L AND BAY 7R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

X

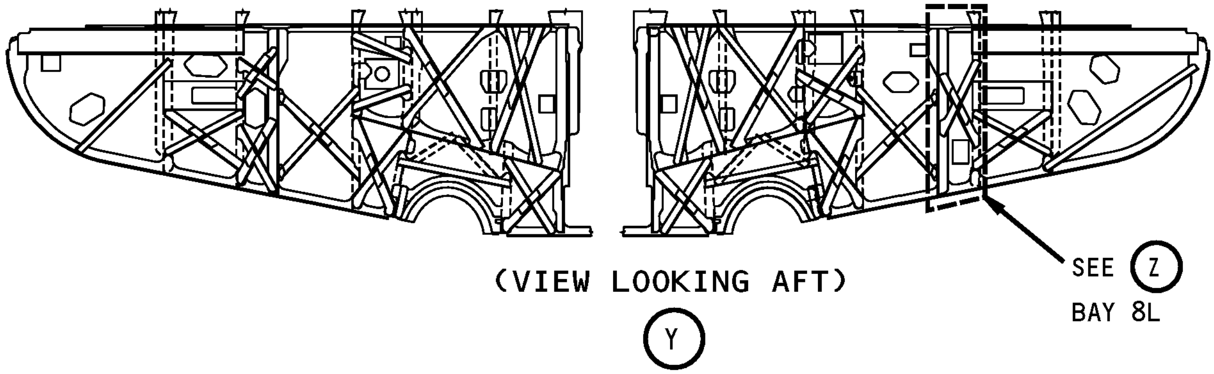
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 24 of 43)**

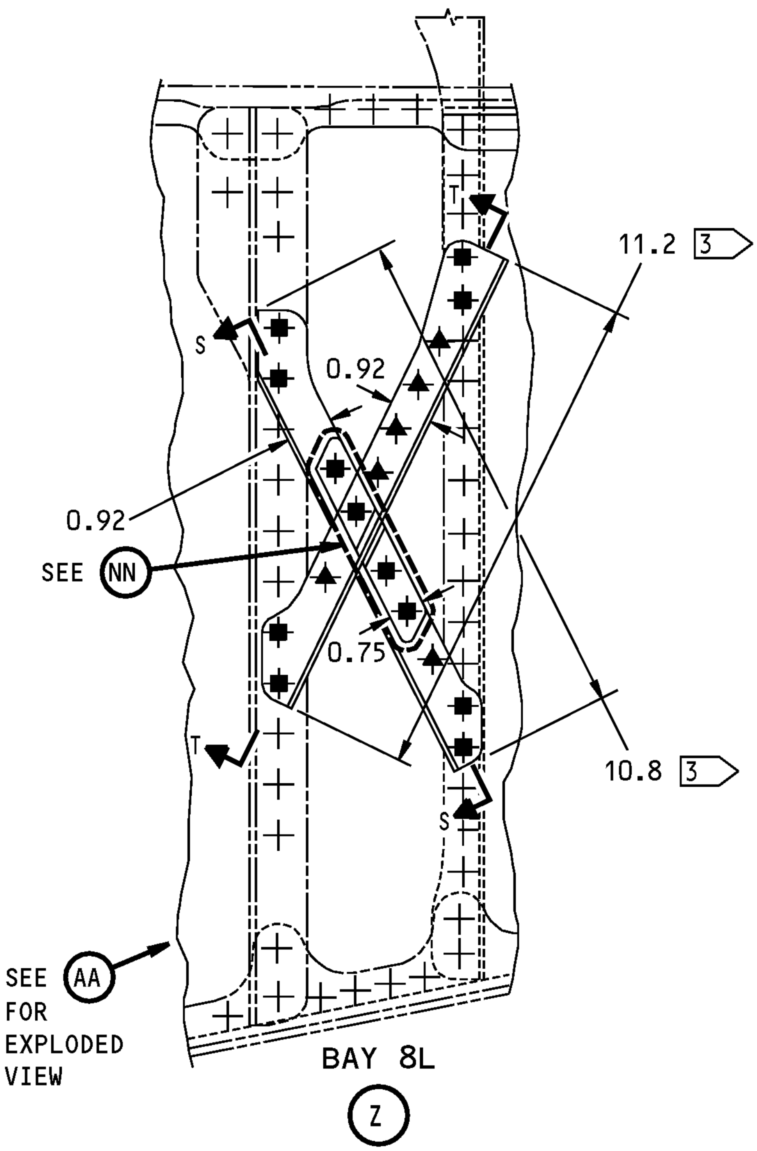



NOTE: REFER TO THE TABLE 209 FOR BAY 7L AND BAY 7R MATERIALS

J86704 S0000181689_V5

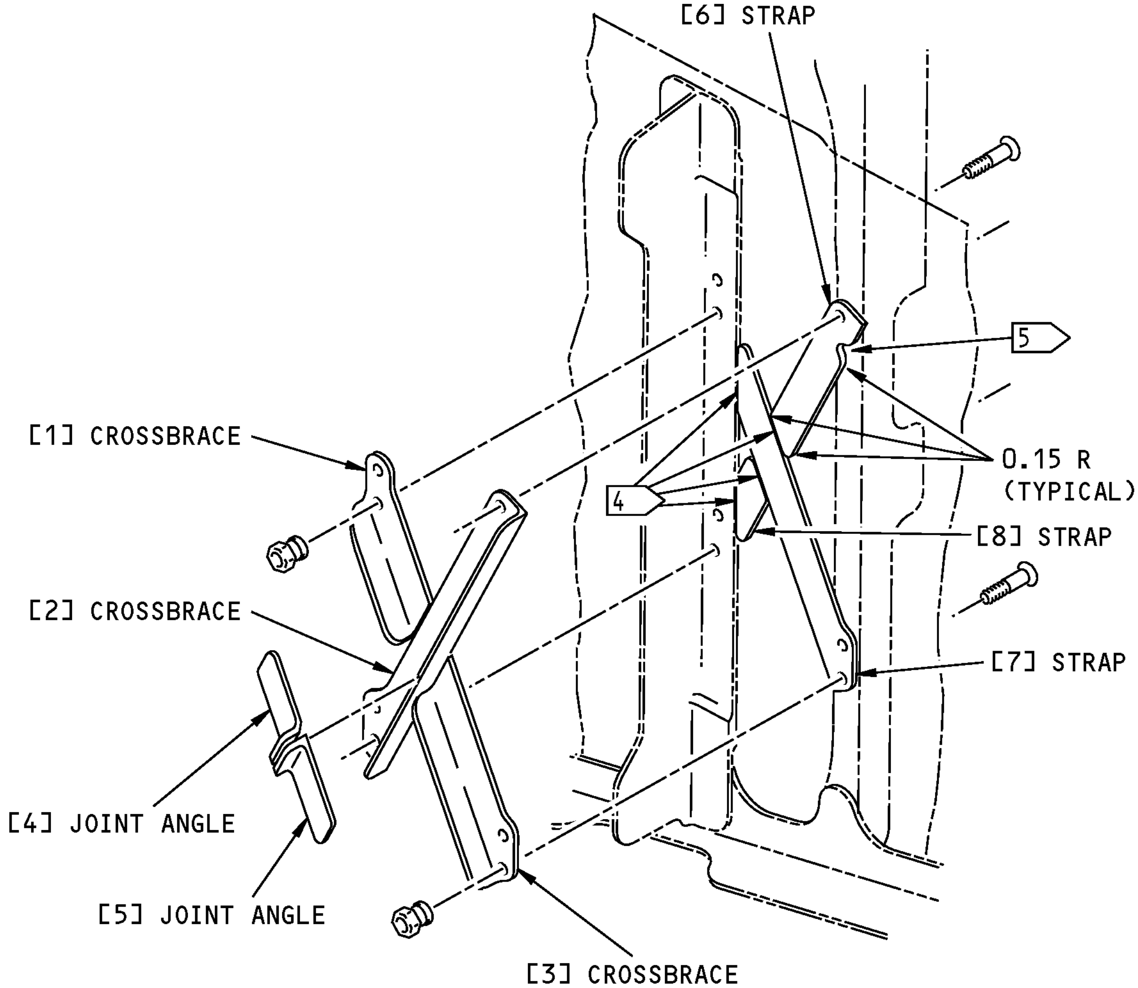
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 25 of 43)**




NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 26 of 43)**

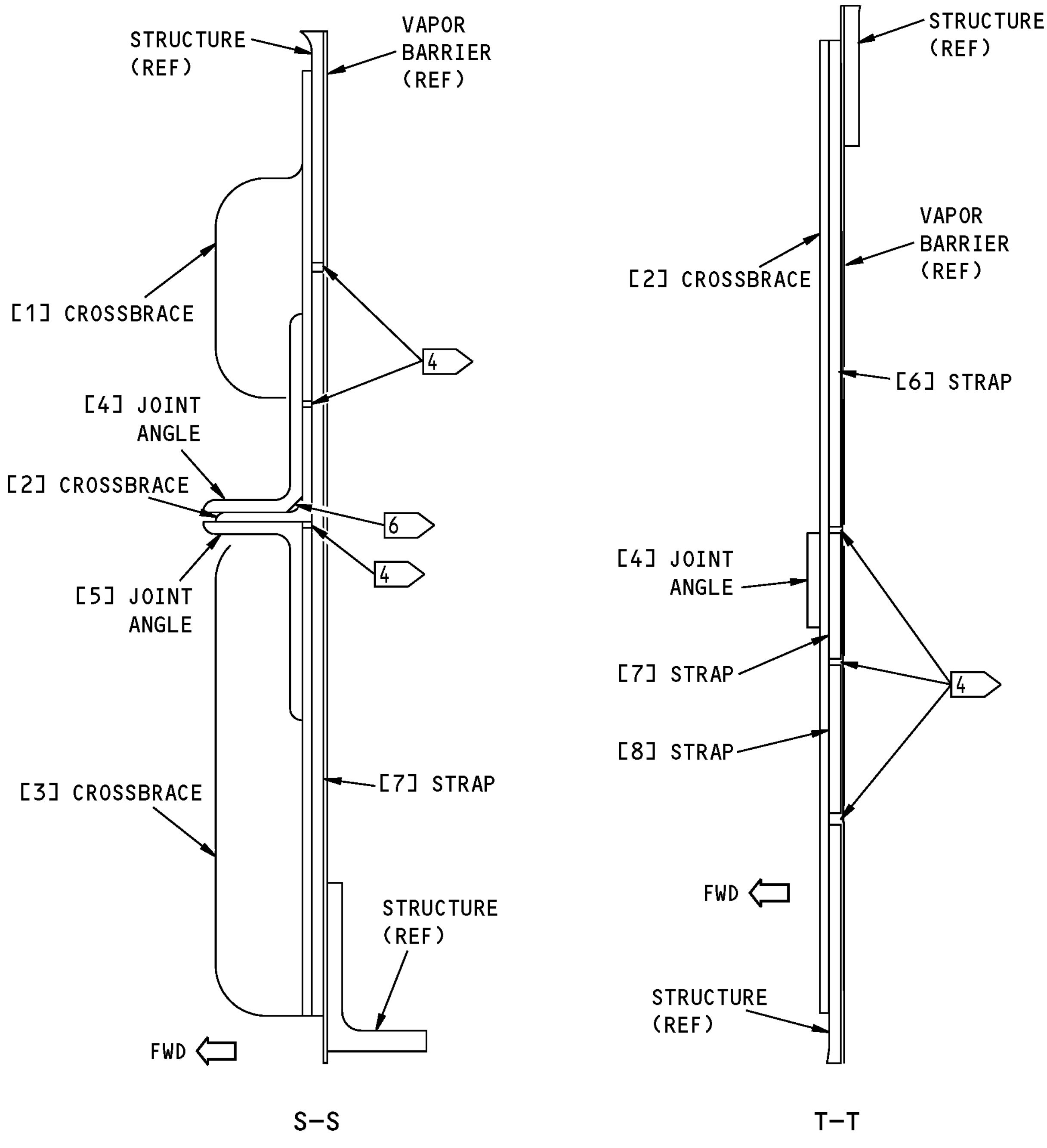



NOTE: REFER TO THE TABLE 210 FOR BAY 8L MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

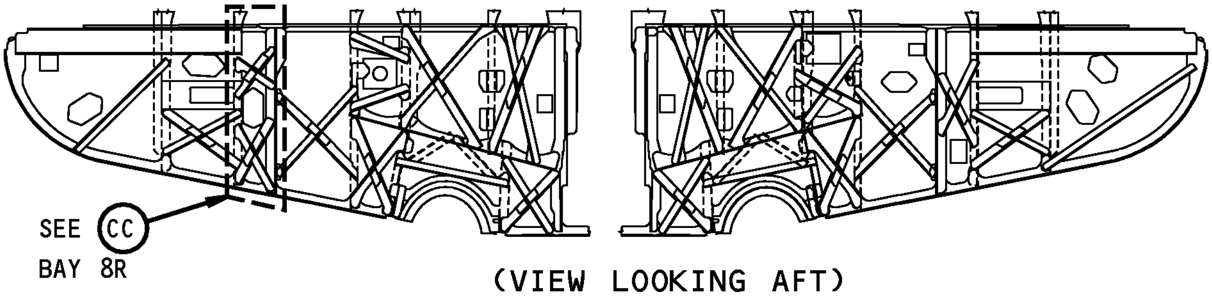
AA

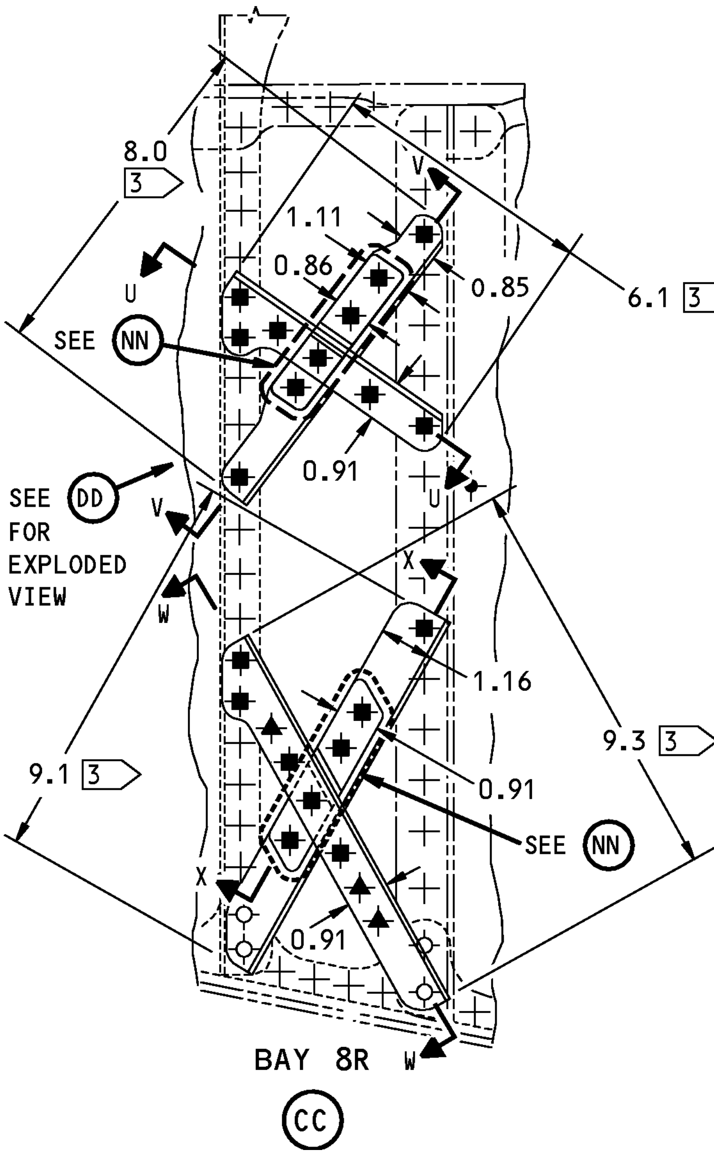
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 27 of 43)**




NOTE: REFER TO THE TABLE 210 FOR BAY 8L MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 28 of 43)**

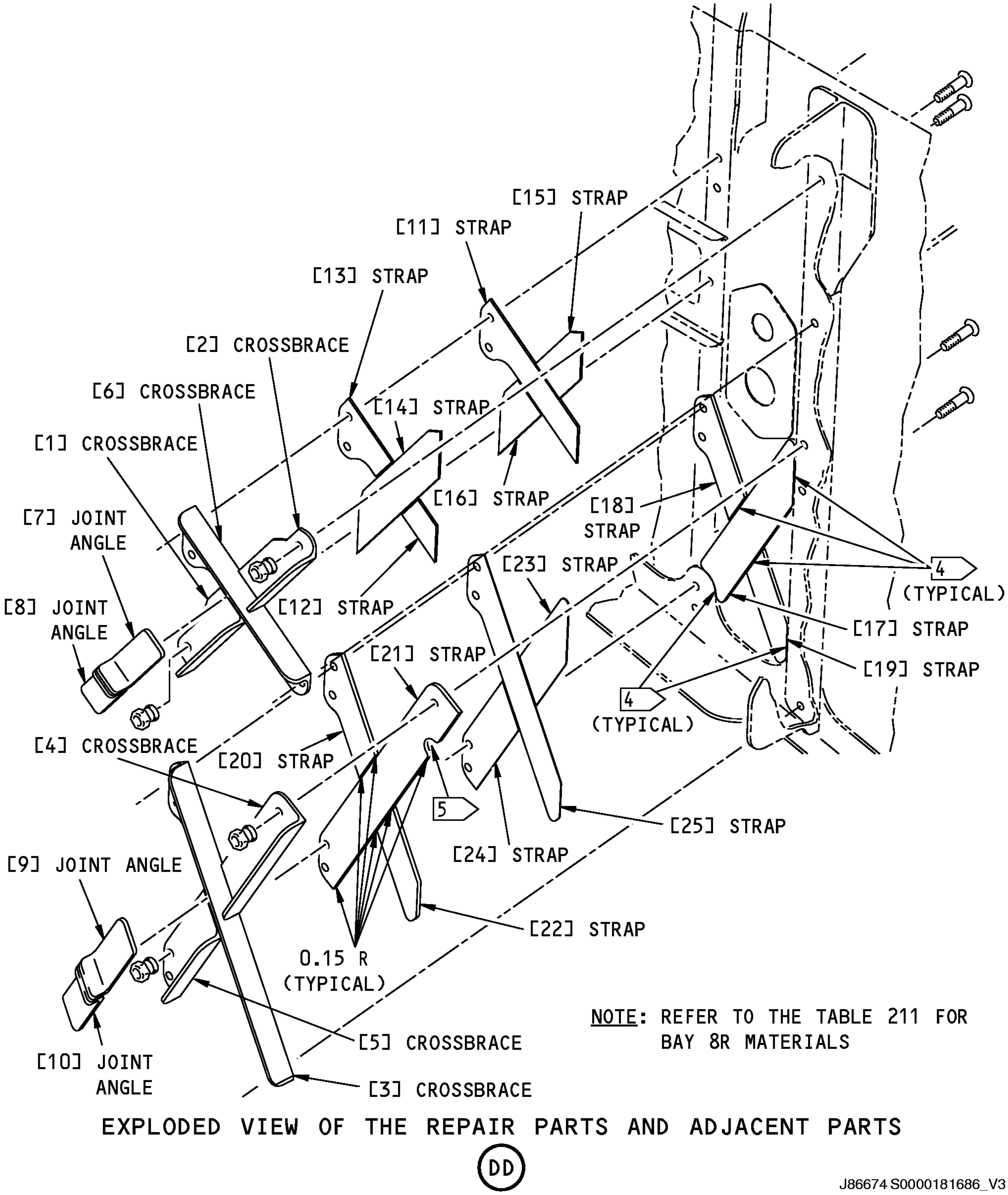


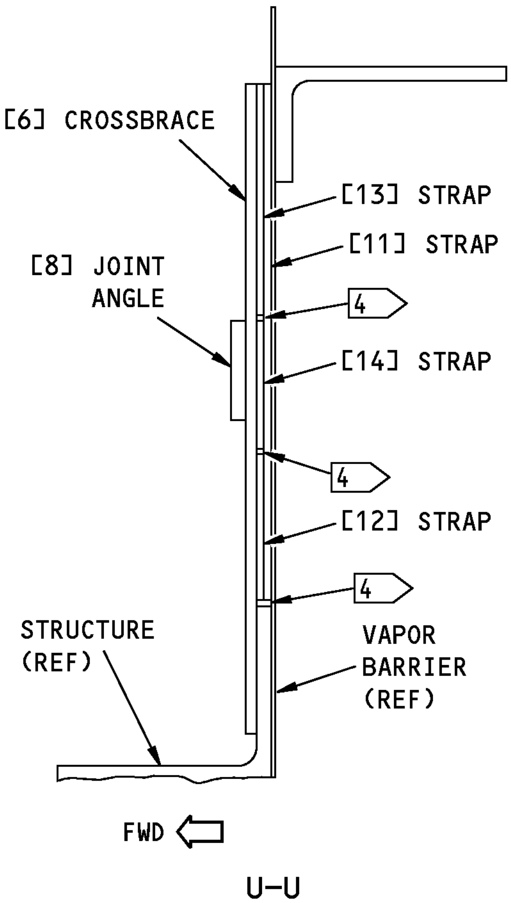

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

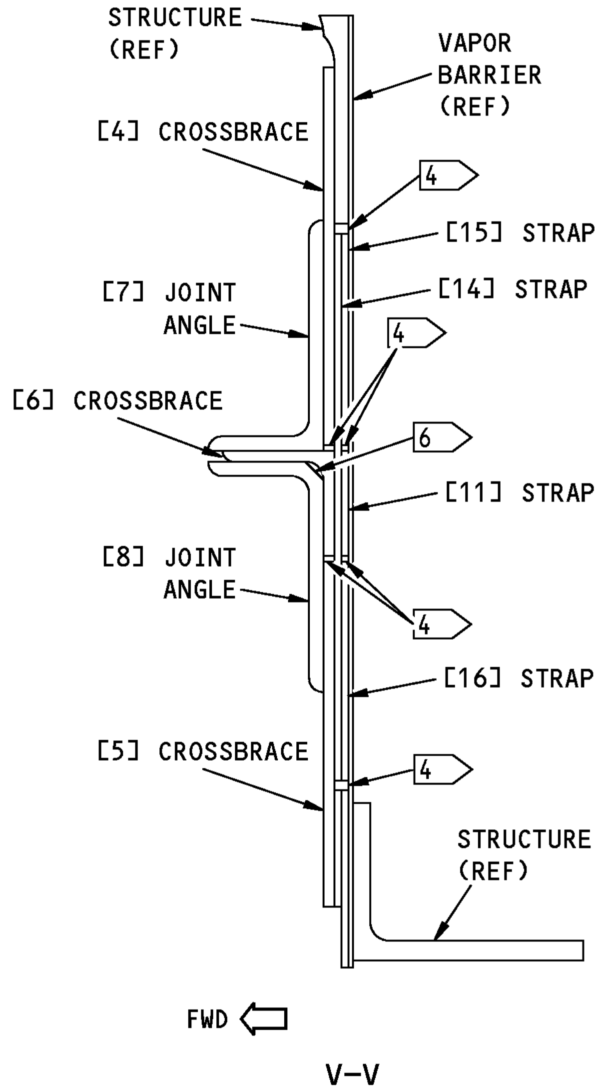
J85782 S0000181666_V2

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 29 of 43)**




**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 30 of 43)**

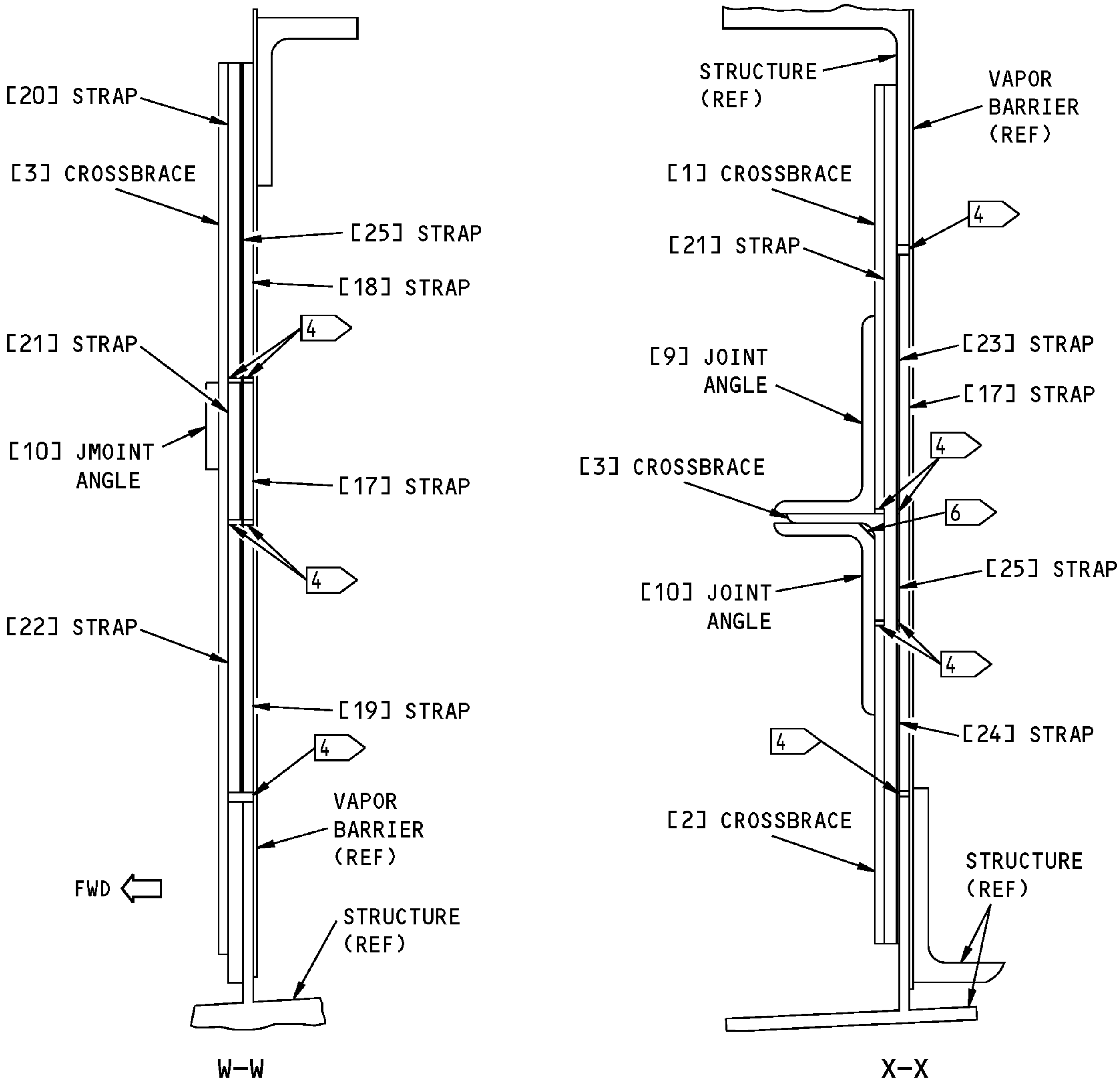



NOTE: REFER TO THE TABLE 211 FOR BAY 8R MATERIALS

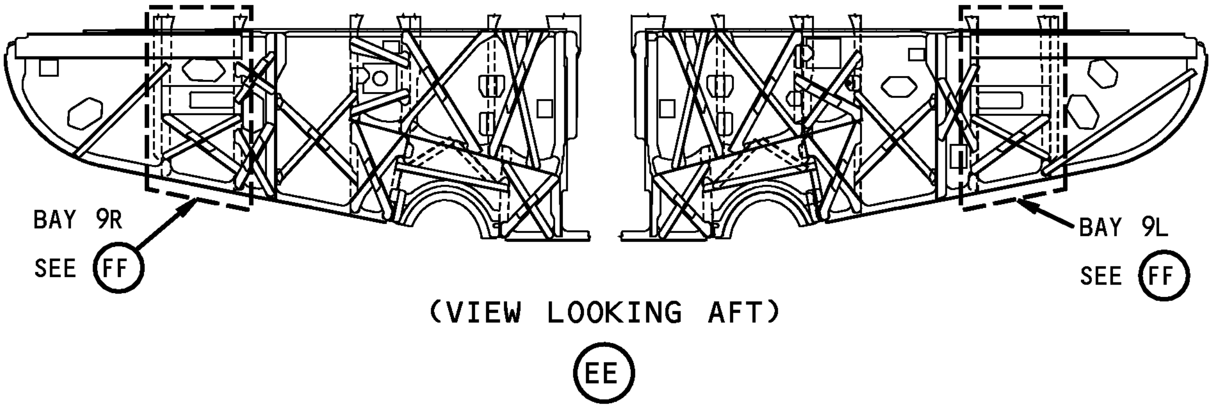
J86868 S0000181691_V3

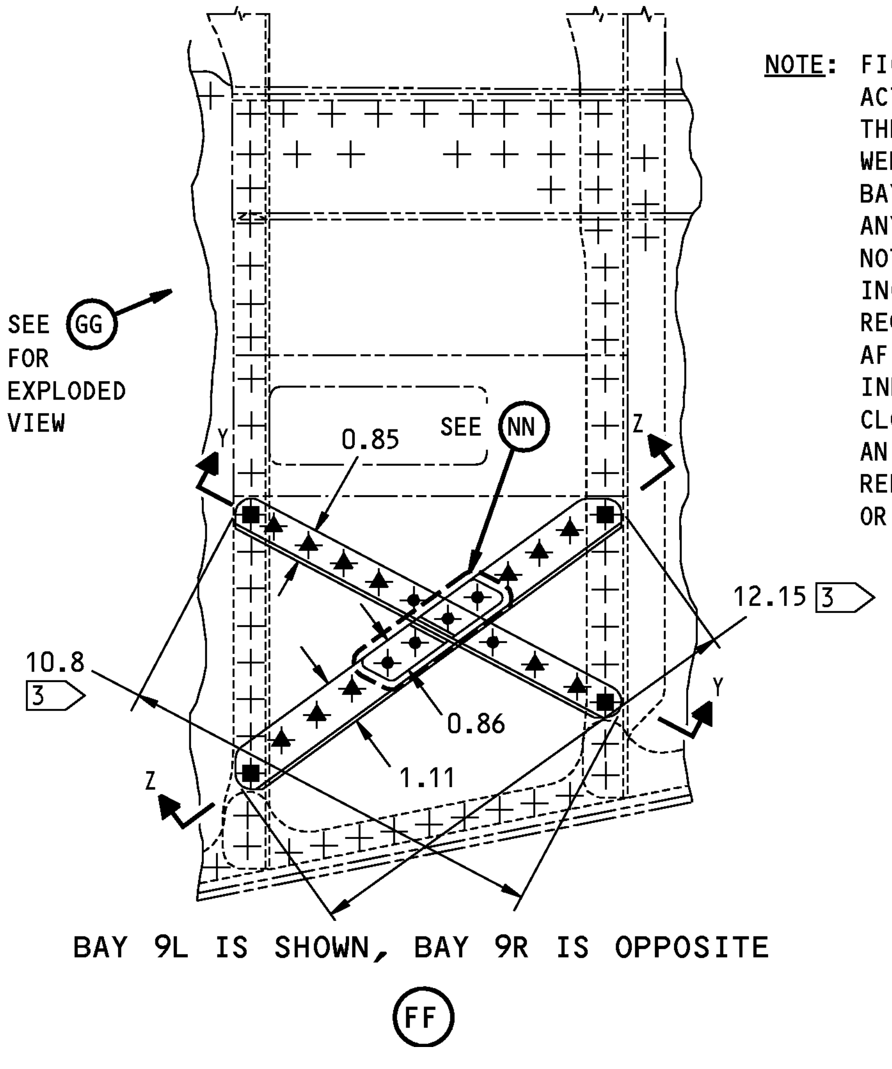
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 31 of 43)**




<u>NOTE</u>: REFER TO THE TABLE 211 FOR BAY 8R MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 32 of 43)**

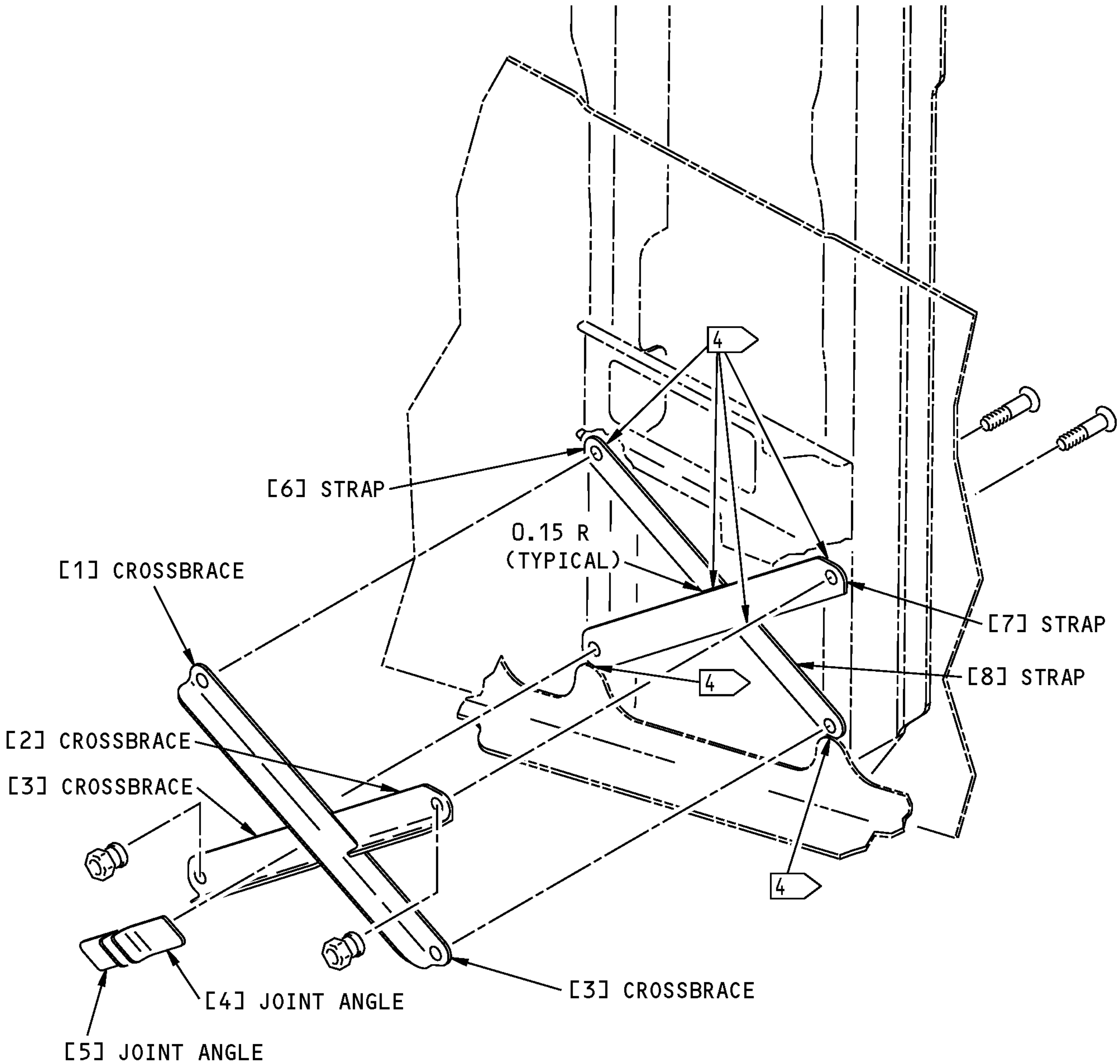



NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85834 S0000181668_V3

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 33 of 43)**

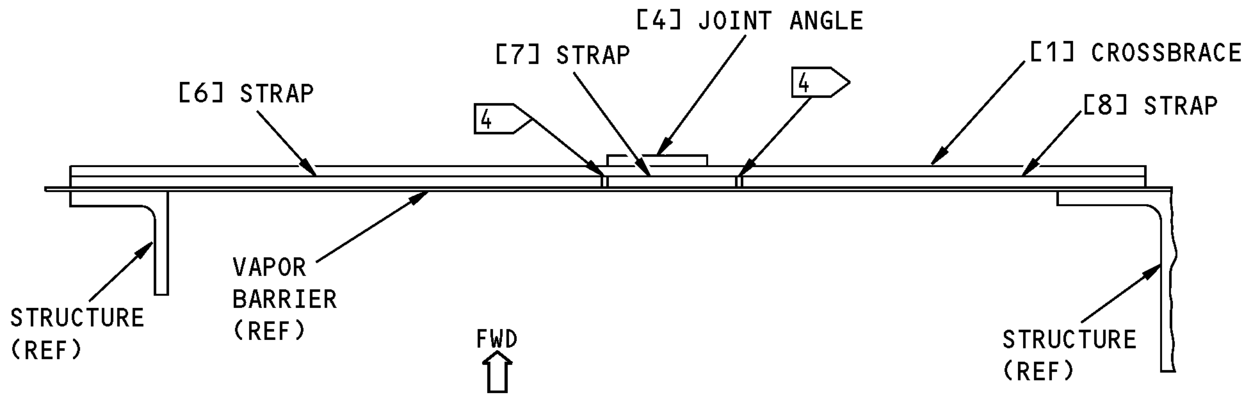



NOTE: REFER TO THE TABLE 212 FOR BAY 9L AND BAY 9R MATERIALS

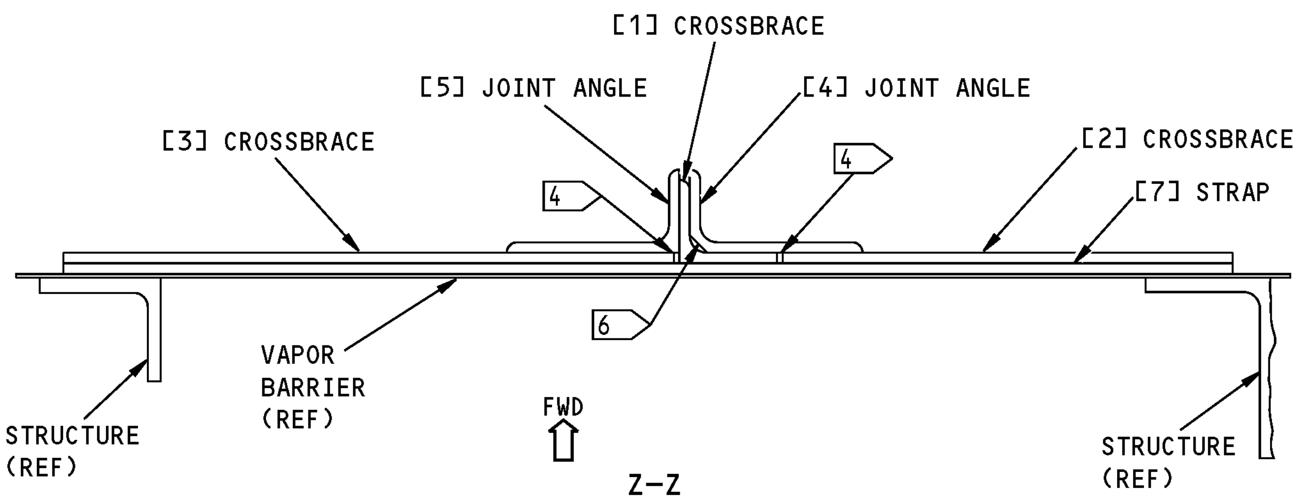
EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

GG

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 34 of 43)**



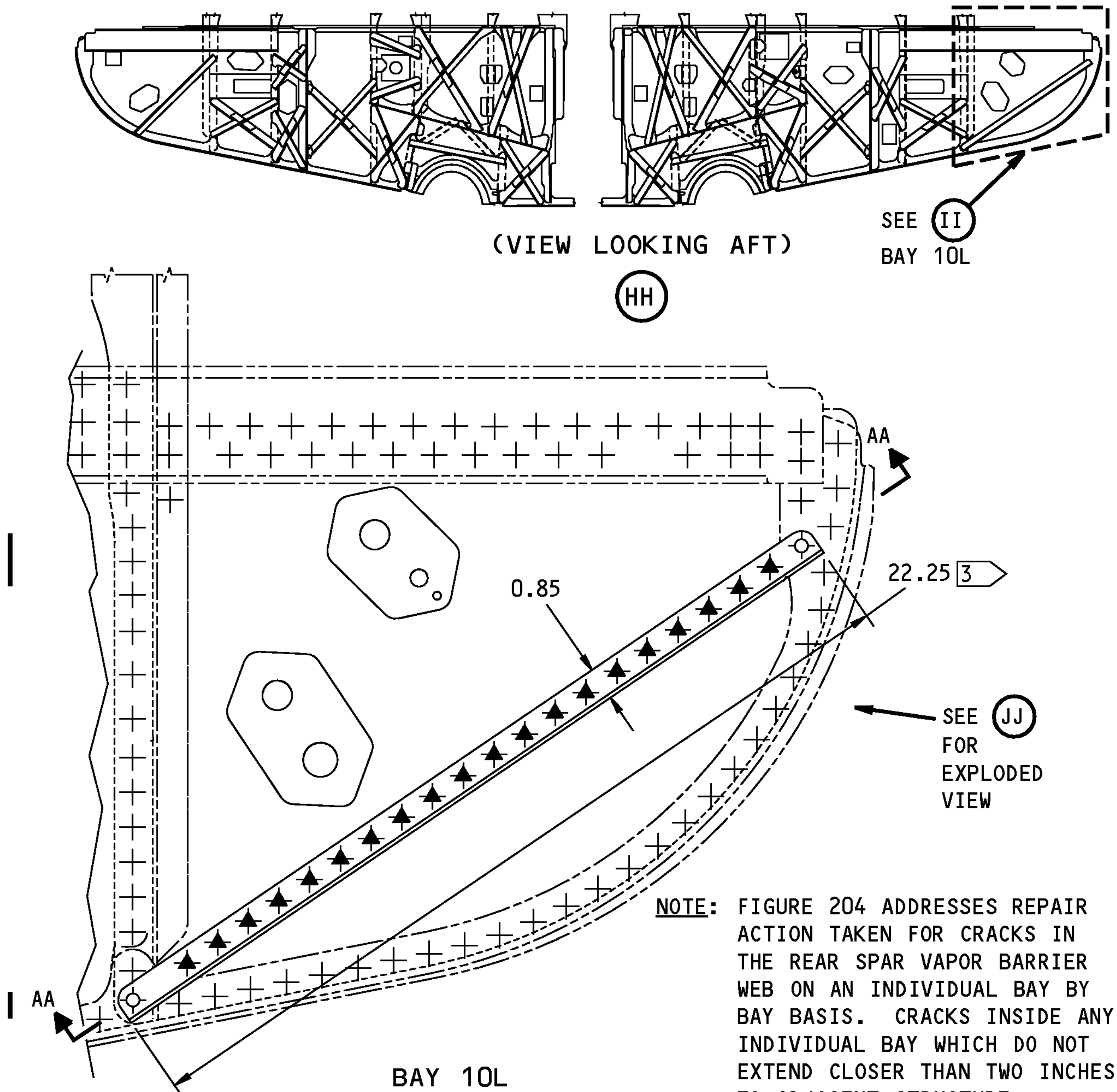

Y-Y




Z-Z

NOTE: REFER TO THE TABLE 212 FOR BAY 9L
AND 9R MATERIALS

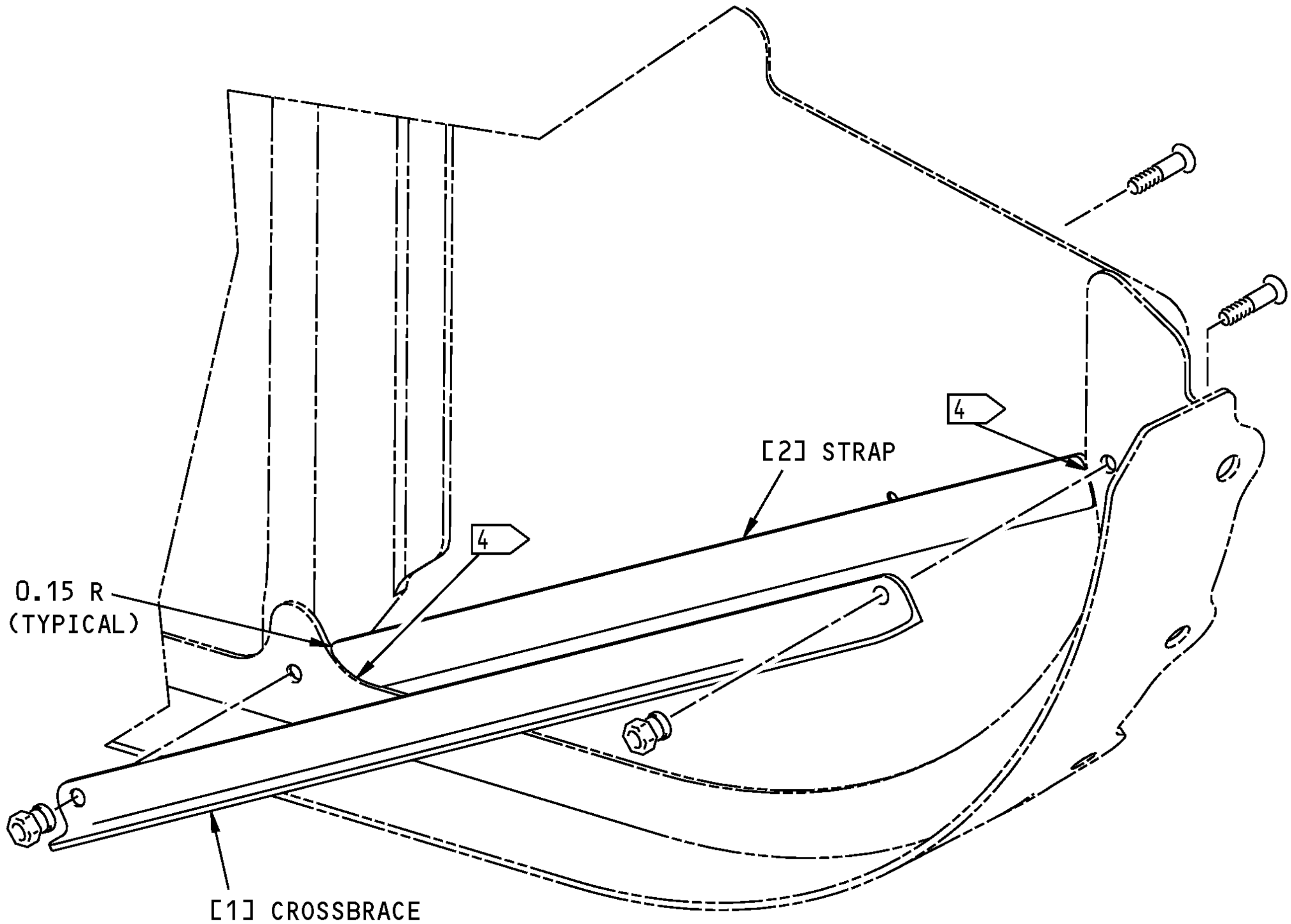
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 35 of 43)**




NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

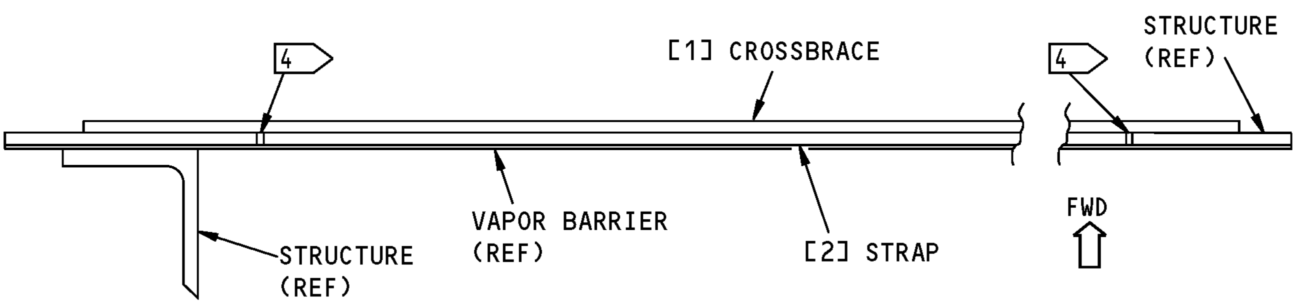
J85843 S0000181671_V2

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 36 of 43)**




EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

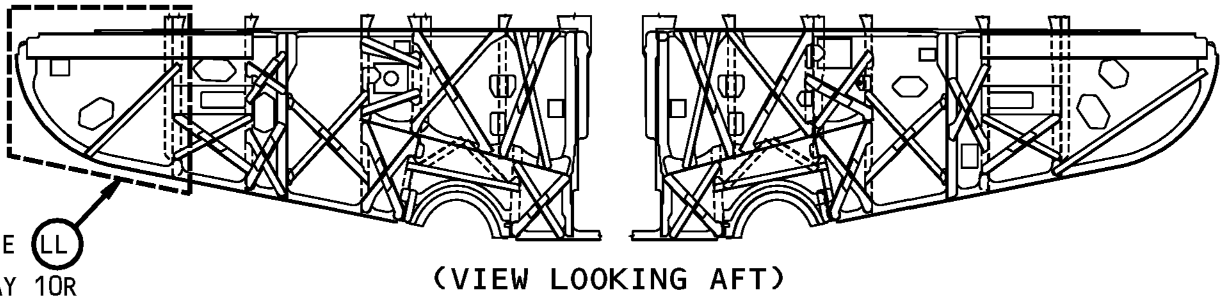
JJ

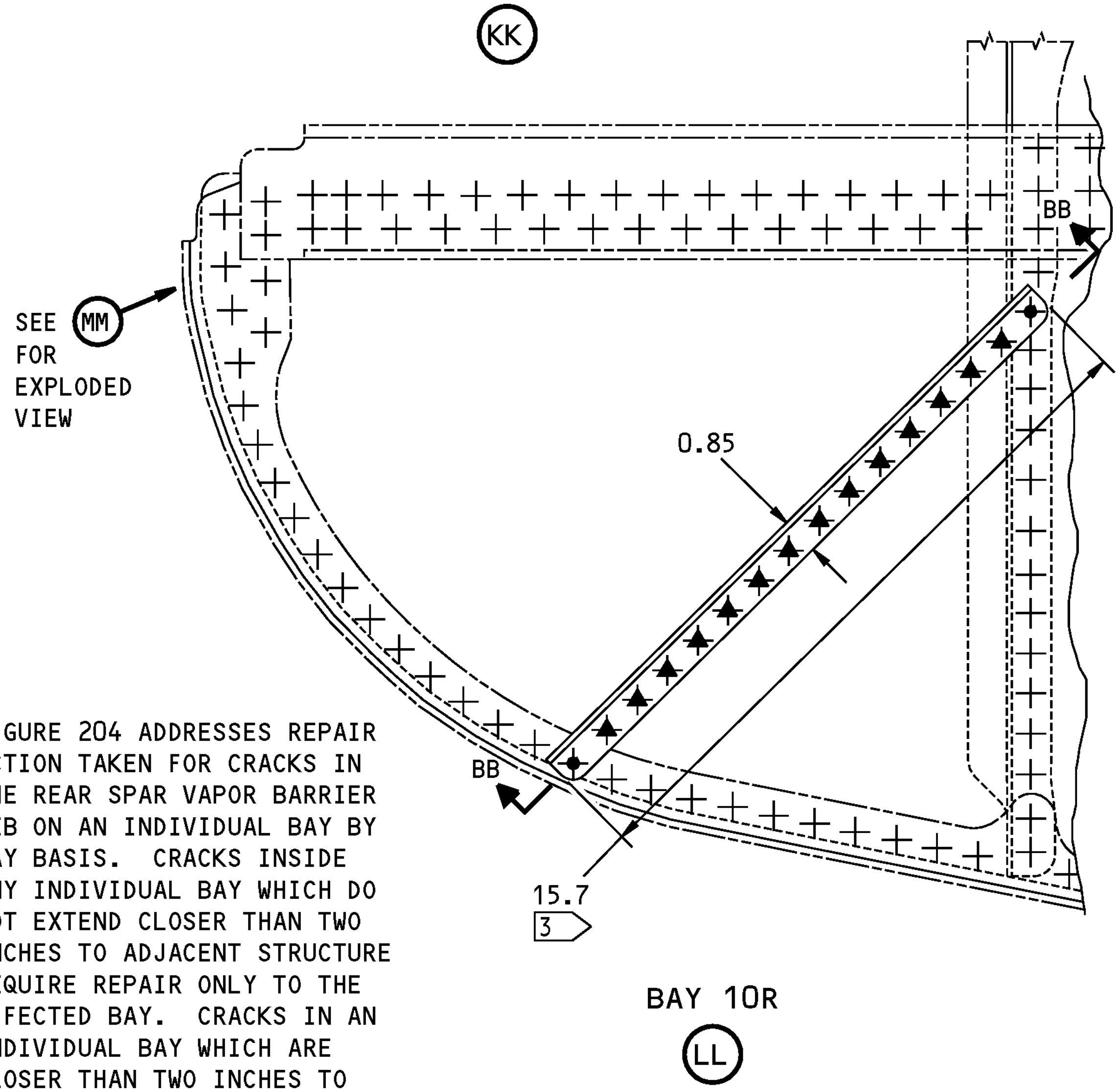



AA-AA

NOTE: REFER TO THE TABLE 213 FOR BAY 10L MATERIALS

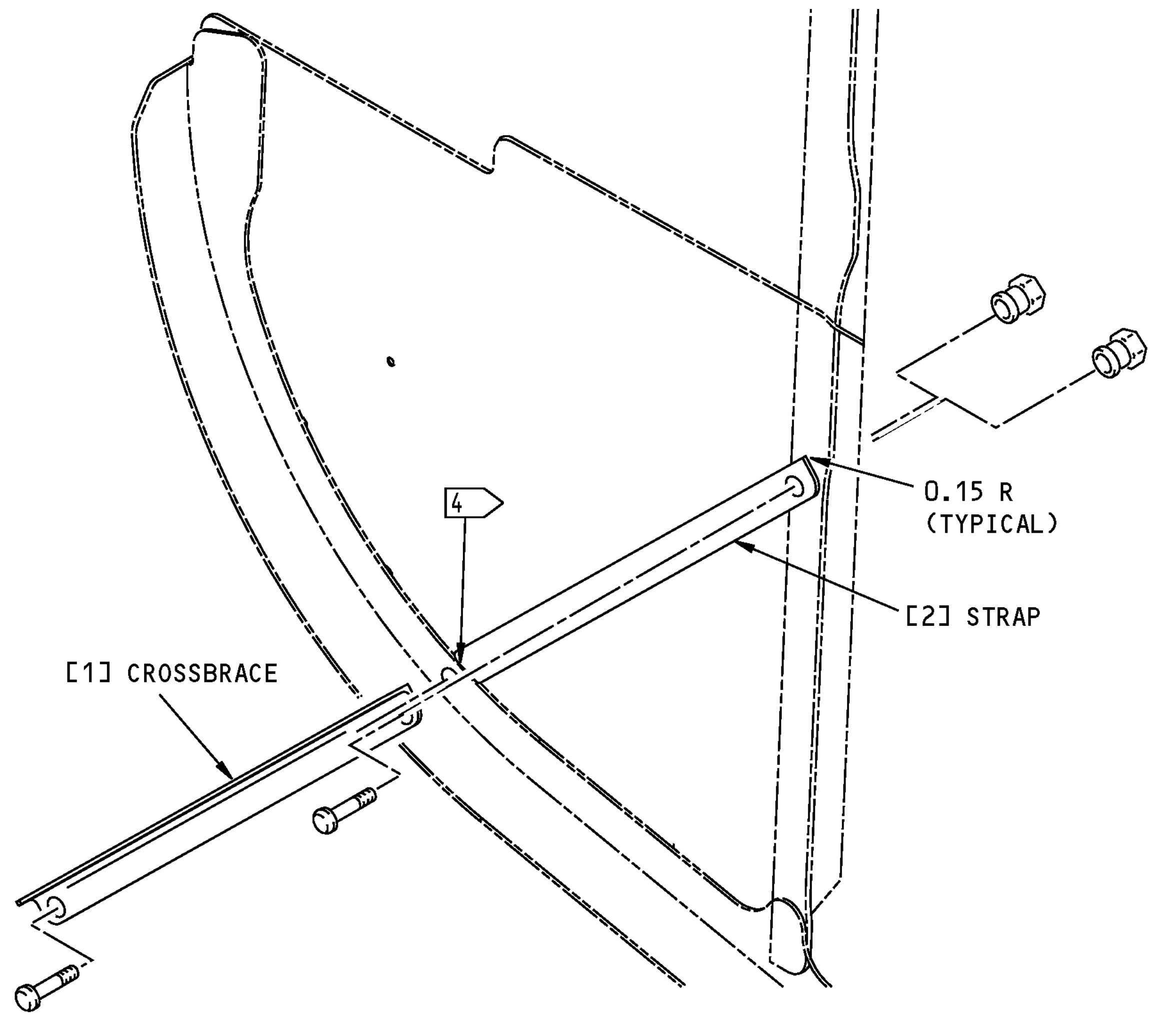
**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 37 of 43)**



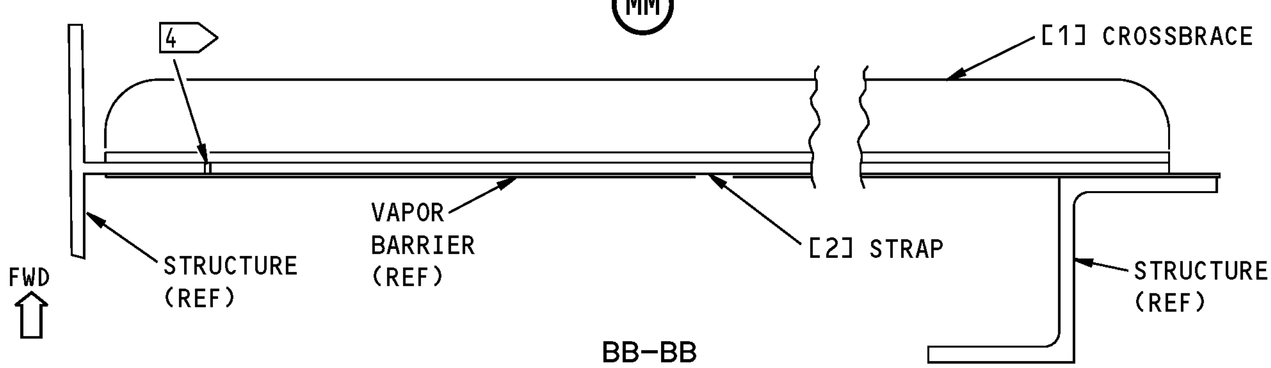

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 38 of 43)**




EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

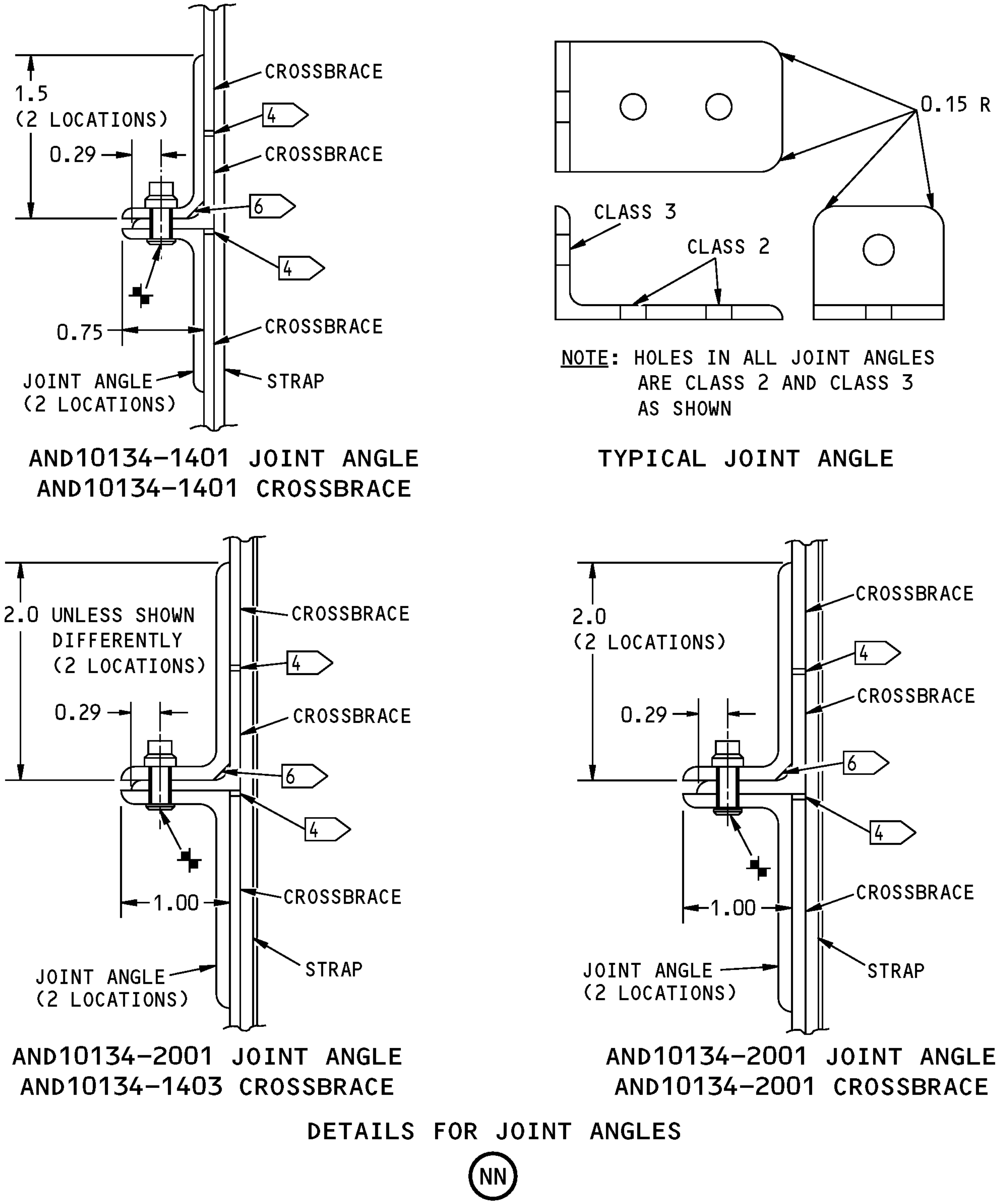
MM



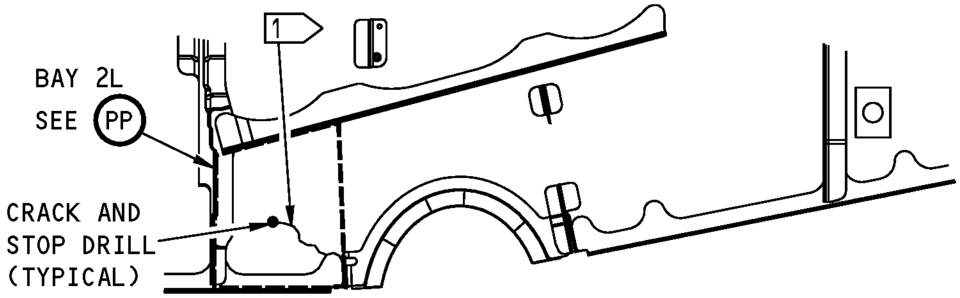

BB-BB

NOTE: REFER TO THE TABLE 214 FOR BAY 10R MATERIALS

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 39 of 43)**



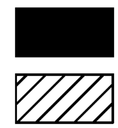

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 40 of 43)**




FILLET SEAL (BMS 5-95), PRIMER
(BMS 10-11, TYPE 1), AND BMS 5-81
FOR SINGLE BAY CROSSBRACE REPAIR
(VIEW LOOKING AFT)

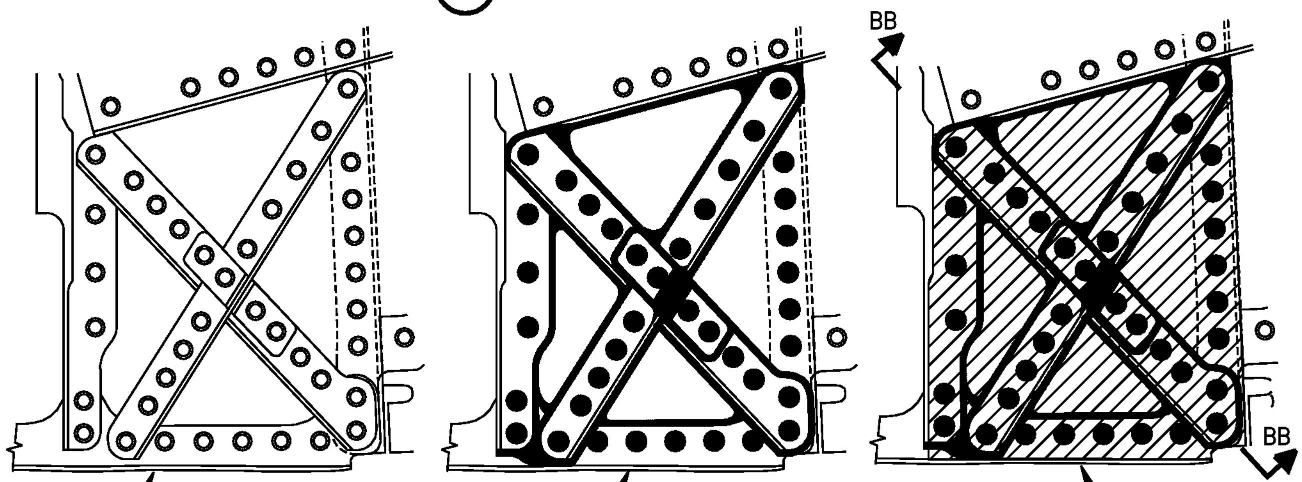
OO

LEGEND:



BMS 5-95

BMS 10-11,
TYPE 1 AND
BMS 5-81

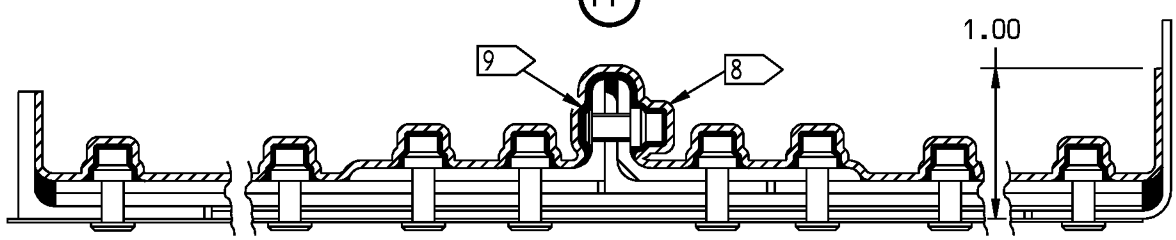



JOIN THE NEW FABRICATED
PARTS TO THE BAY

APPLY FILLET SEAL (BMS 5-95) TO
ALL THE EDGES OF REPAIR PARTS,
SURROUNDING PARTS, AND AFFECTED
FASTENERS BEFORE APPLICATION OF
BMS 10-11, TYPE 1, PRIMER AND
BMS 5-81 COATING

APPLY BMS 10-11,
TYPE 1, PRIMER AND
BMS 5-81 COATING TO
THE ENTIRE FORWARD
SIDE OF THE BAY AREA

PP




BB-BB

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 41 of 43)**

NOTES

- PROVIDE 2D MINIMUM EDGE MARGIN AT ALL LOCATIONS UNLESS SHOWN DIFFERENTLY.
- PROVIDE 4-6D FASTENER SPACING AT NEW LOCATIONS.
- INSTALL HI-LOKS WITH WET BMS 5-95.
- ALL BEND RADII TO BE 0.38 TO 0.50 UNLESS SHOWN DIFFERENTLY.
- PARTS TO BE TRIMMED WITH THE DIMENSIONS AND A TOLERANCE OF -0.000 TO +0.050.
- ALL DIMENSIONS ARE IN INCHES.

1 PERMITTED CRACKS ARE MORE THAN 2.0 INCHES FROM AN ADJACENT BAY

2 IF REPAIRED BEFORE AS GIVEN IN FIGURE 203, THEN CONTACT THE BOEING COMPANY FOR INSTRUCTIONS

3 USE THIS DIMENSION TO ORDER THE MATERIAL AND FABRICATE THE PART (DIMENSION INCLUDES EXCESS)

4 GAP BETWEEN THE ENDS OF STRAPS AND ADJACENT STRUCTURE OR PARTS TO BE 0.020 TO 0.080

5 MAKE SURE NEW PARTS FIT AROUND FASTENERS AND STRUCTURES WITH A 0.020 TO 0.080 GAP

6 CHAMFER ANGLE TO CLEAR RADIUS BEFORE PROTECTIVE TREATMENT IS APPLIED (0.120 TO 0.180 BY 45°)

7 RADIUS PARTS ARE 0.12 TO 0.18 R TO FIT INTO RADIUS OF ADJACENT PART

8 CAUTION: THE BMS 5-81 COATING MAKES A FLAMMABLE VAPOR BARRIER. MAKE SURE THE VAPOR BARRIER IS NOT DAMAGED.

APPLY BMS 10-11, TYPE 1, PRIMER AND BMS 5-81 COATING AS SHOWN IN BAC 5527 OVER FORWARD SURFACE OF VAPOR BARRIER WEB ASSEMBLY AND ALL SURFACES OF NEWLY ATTACHED PARTS TO A THICKNESS OF 0.015 TO 0.025. BMS 5-81 COATING IS TO EXTEND 1.0 INCH MINIMUM ONTO SURFACES OF COMPONENTS ATTACHED TO THE VAPOR BARRIER WEB

9 FILLET SEAL (BMS 5-95) ALL EDGES OF REPAIR PARTS, SURROUNDING PARTS, AND AFFECTED FASTENERS ON THE FORWARD SIDE OF THE VAPOR BARRIER WEB ASSEMBLY

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 42 of 43)**

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL AND REPAIR FASTENER LOCATIONS:
- HOLE; 0.1635 TO 0.1645 DIA
- EDGE MARGIN; 0.3270 MIN
- HEX DRIVE BOLT; BACB30VT5
- COLLAR; BACC30BL5

REPAIR FASTENER LOCATION:
- HOLE; 0.1660 TO 0.1710 DIA
- EDGE MARGIN; 0.3320 MIN
- HEX DRIVE BOLT; BACB30VT5HK()
- COLLAR; BACC30BL5

INITIAL AND REPAIR FASTENER LOCATIONS:
- HOLE; 0.1895 TO 0.1905 DIA
- EDGE MARGIN; 0.3790 MIN
- HEX DRIVE BOLT; BACB30VT6K
- COLLAR; BACC30BL6

INITIAL FASTENER LOCATION:
- HOLE; 0.2495 TO 0.2505 DIA
- EDGE MARGIN; 0.4990 MIN
- HEX DRIVE BOLT; BACB30VT8K
- COLLAR; BACC30BL8

INITIAL AND REPAIR FASTENER LOCATIONS:
- HOLE; 0.1635 TO 0.1645 DIA
- EDGE MARGIN; 0.3270 MIN
- HEX DRIVE BOLT; BACB30MB5
- COLLAR; BACC30BH5

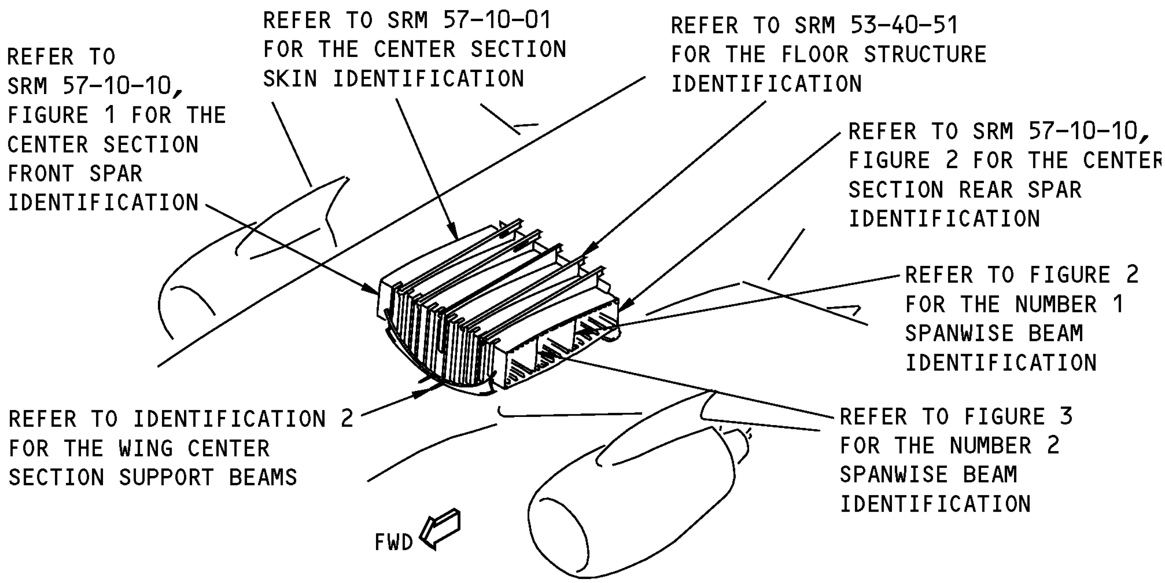
INITIAL FASTENER LOCATION:
- HOLE; 0.1895 TO 0.1905 DIA
- EDGE MARGIN; 0.3790 MIN
- HEX DRIVE BOLT; BACB30MB6
- COLLAR; BACC30BH6

REPAIR FASTENER LOCATION:
- HOLE; 0.1920 TO 0.2030 DIA
- EDGE MARGIN; 0.29 MIN
- HEX DRIVE BOLT; BACB30VT5HK()
- COLLAR; BACC30BL5

J88485 S0000184336_V3

**Wing Center Section Rear Spar Web Cross Brace Repair**
**Figure 204 (Sheet 43 of 43)**

## IDENTIFICATION 1 - WING CENTER SECTION SPANWISE BEAMS

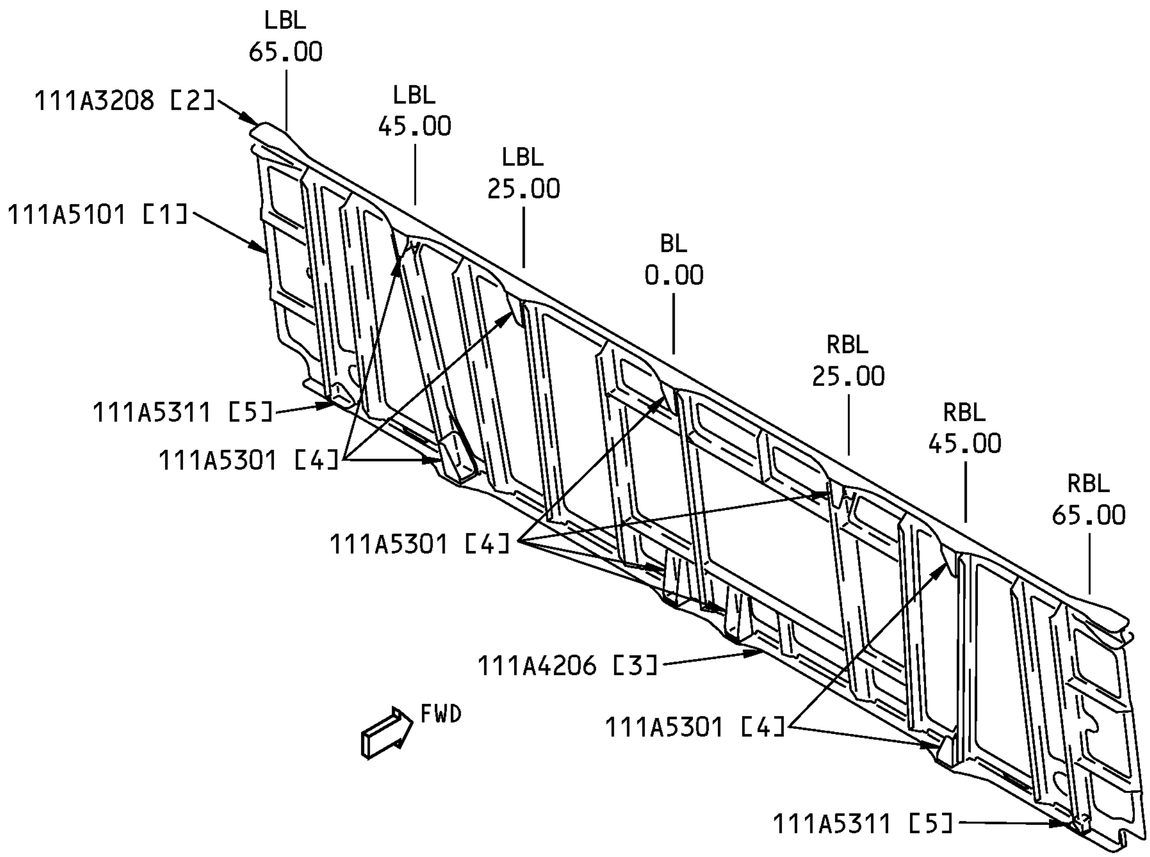



NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Center Section Spanwise Beam Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 111A5000 | Spanwise Beam Installation - Wing Center Section |
| 111A5001 | Spanwise Beam Assembly - No. 1, Wing Center Section |
| 111A5002 | Spanwise Beam Assembly - No. 2, Wing Center Section |



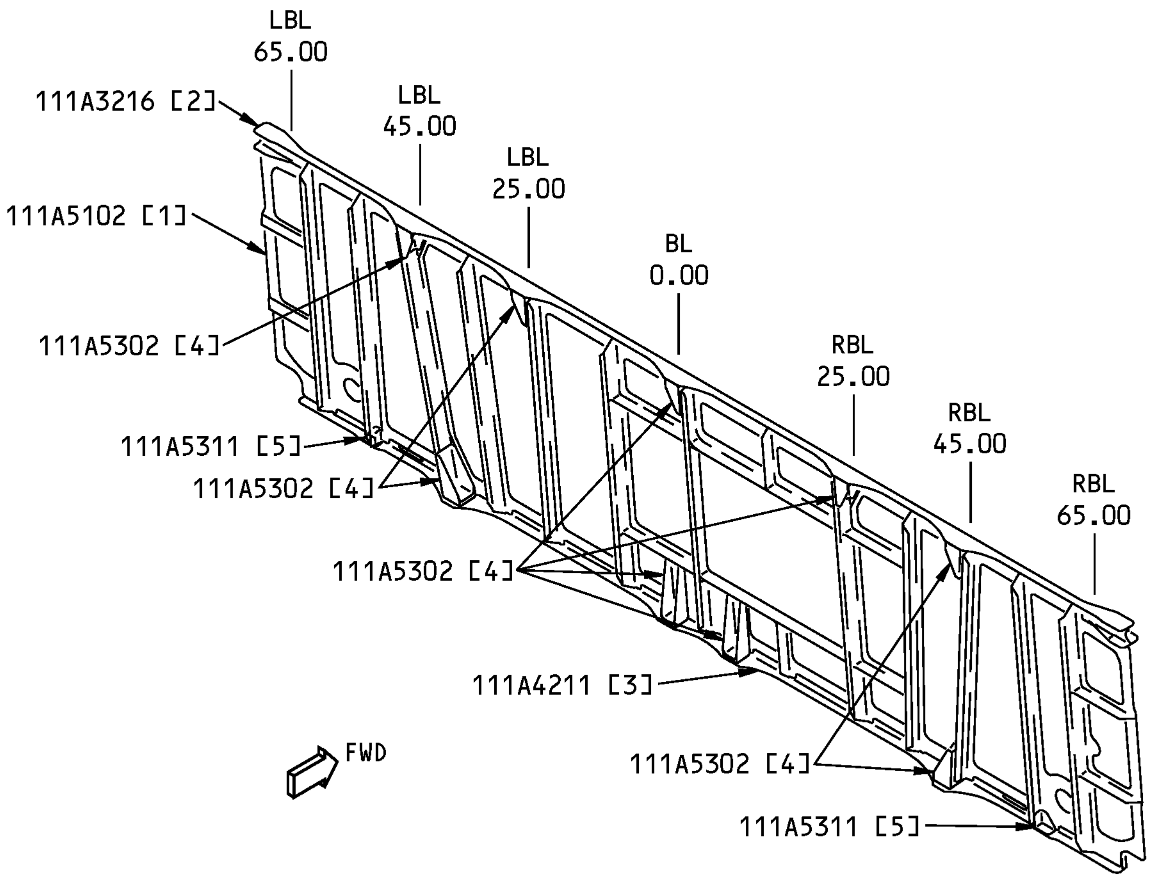

NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Wing Center Section Spanwise Beam Number 1 Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Integral Web Number 1 | | 7050-T7451 plate that is machined to different thicknesses | |
| [2] | Number 8 Stringer | | BAC1518-1168 7055-T77511 extrusion as given in BMS 7-308 | Line Number 1 thru 1537 |
| [2] | Number 8 Stringer | | BAC1518-1266 7055-T77511 extrusion as given in BMS 7-308 | Line Number 1538 and on |
| [3] | Number 6 Stringer | | BAC1517-2739 2224-T3511 extrusion as given in BMS 7-255 | |
| [4] | Tension Fitting | | 7050-T7451 plate | |
| [5] | Backup Fitting | | 7050-T7451 plate | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

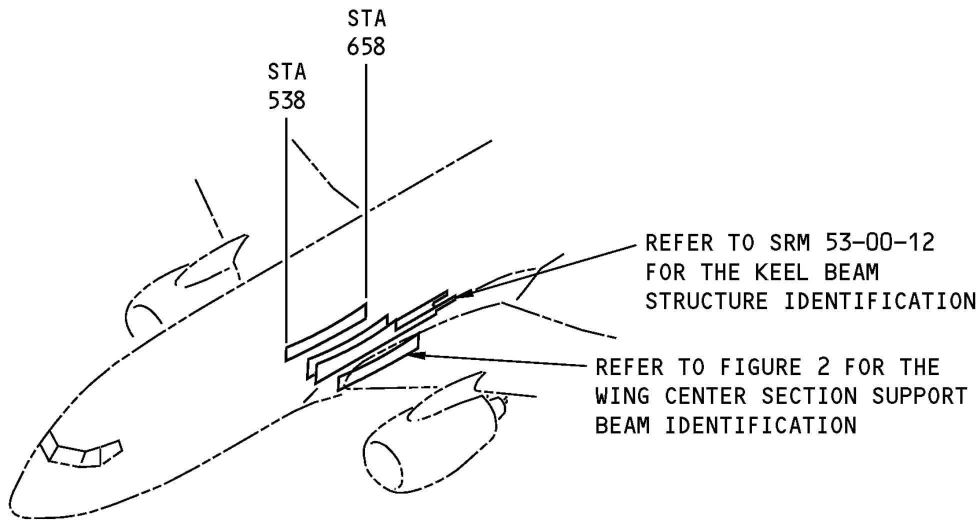
**Wing Center Section Spanwise Beam Number 2 Identification**
**Figure 3**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Integral Web Number 2 | | 7050-T7451 plate that is machined to different thicknesses | |
| [2] | Number 16 Stringer | | BAC1518-1168 7055-T77511 extrusion as given in BMS 7-308 | |
| [3] | Number 11 Stringer | | BAC1517-2738 2224-T3511 extrusion as given in BMS 7-255 | |
| [4] | Tension Fitting | | 7050-T7451 plate | |
| [5] | Backup Fitting | | 7050-T7451 plate | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - WING CENTER SECTION SUPPORT BEAMS

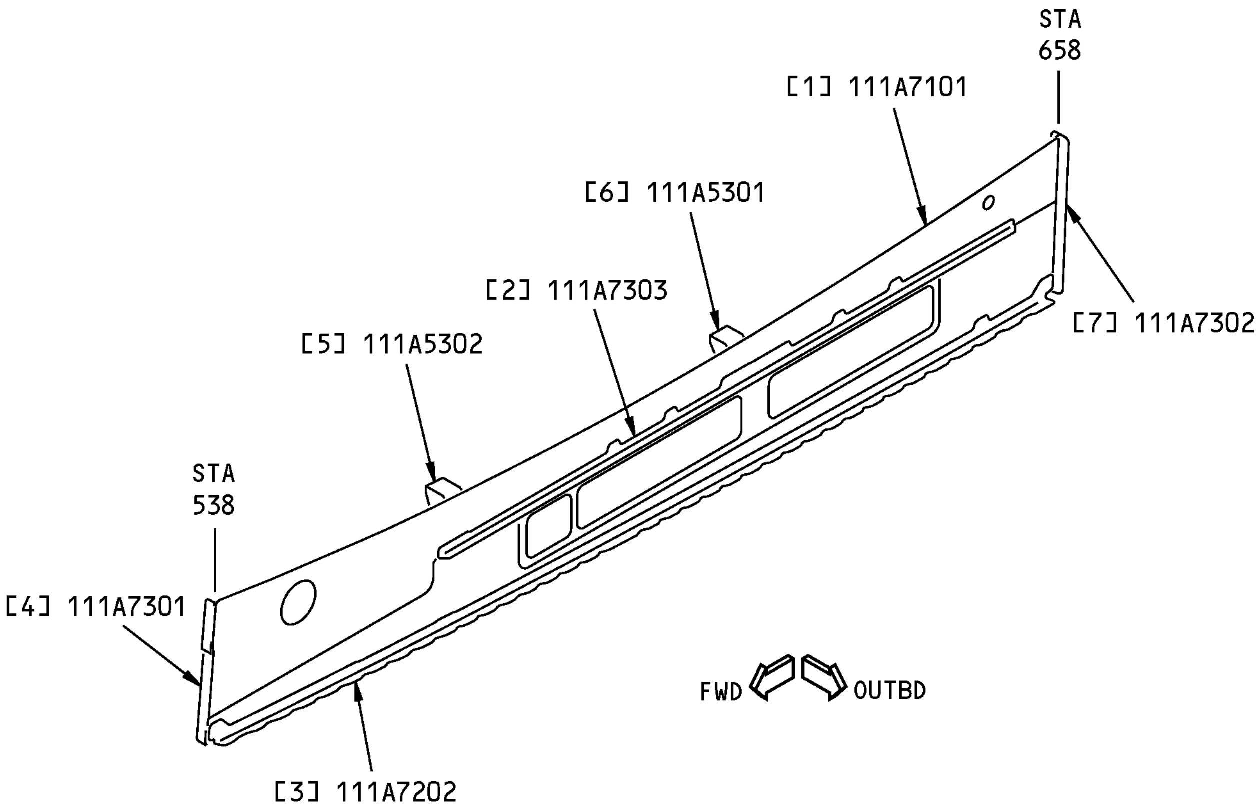



NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Center Section Support Beam Locations**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 111A7000 | Lower Beam Installation - Wing Center Section |
| 111A7001 | Lower Beam Assembly - Wing Center Section |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE SUPPORT BEAM IS SHOWN,
RIGHT SIDE SUPPORT BEAM IS OPPOSITE

**Wing Center Section Support Beam Identification**
**Figure 2**

**Table 2**:

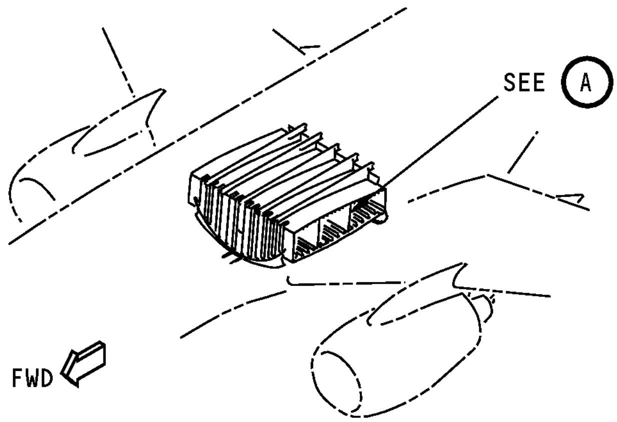
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Lower Beam Web | | 7050-T7451 plate that is machined to different thicknesses | |
| [2] | Horizontal Stiffener | | BAC1506-4365 7075-T3511 extrusion | |
| [3] | Lower Chord | | BAC1506-4274 7075-T73 extrusion | |
| [4] | Front Spar Attach Stiffener | | BAC1514-2763 7075-T3511 extrusion | |
| [5] | Tension Fitting | | 7050-T7451 plate | |
| [6] | Tension Fitting | | 7050-T7451 plate | |
| [7] | Rear Spar Attach Stiffener | | BAC1505-101665 7075-T3511 extrusion | |

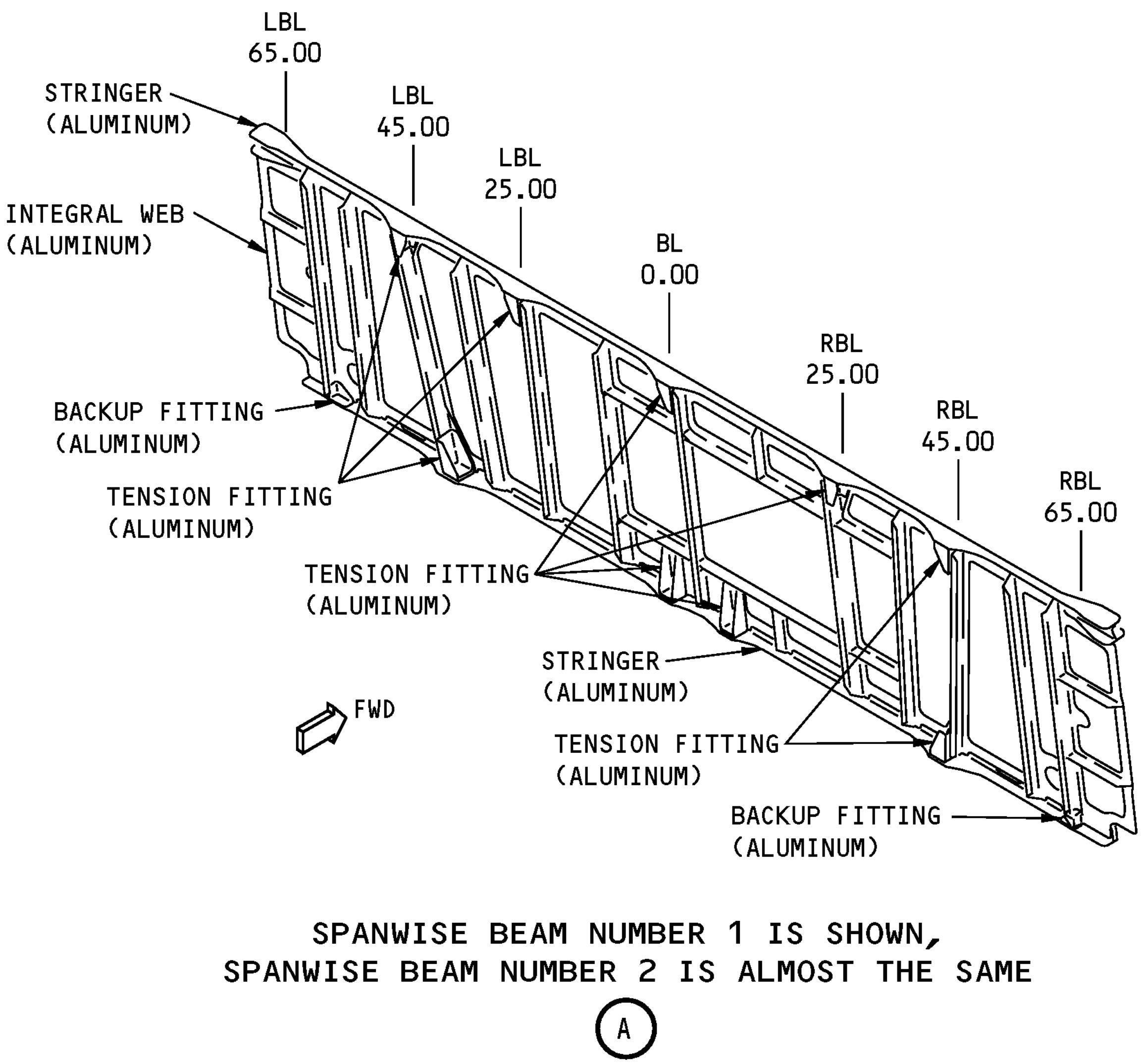
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## ALLOWABLE DAMAGE 1 - WING CENTER SECTION SPANWISE BEAMS

1. **Applicability**

   A. This subject is applicable to the wing center section spanwise beams. Refer to Wing Center Section Spanwise Beam Allowable Damage, Figure 101/ALLOWABLE DAMAGE 1.




**Wing Center Section Spanwise Beam Allowable Damage**
**Figure 101**

2. **General**

**WARNING:** THE GASSES FROM THE FUEL IN THE FUEL TANK ARE DANGEROUS AND EXPLOSIVE. MAKE SURE YOU REMOVE ALL OF THE FUEL FROM THE FUEL TANKS. MAKE SURE THAT THE FUEL TANKS HAVE A GOOD FLOW OF AIR. USE RESPIRATORY EQUIPMENT AND WEAR PROTECTIVE CLOTHING WHEN YOU GO IN THE FUEL TANKS. DO NOT USE EQUIPMENT THAT IS A SOURCE OF HEAT OR SPARKS. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS AND DAMAGE TO THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

C. Remove the damage material as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasives and other materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the tools and equipment you can use to remove the damage.

D. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

E. After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the areas that have rework. Refer to 51-20-01.

(3) Apply two layers of BMS 10-20, Type II primer to the areas that have rework. Refer to AMM 28-11-00/701.

3. **References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-05 | FASTENER HOLE SIZES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| SOPM 20-10-03 | General - Shot Peening Procedures |

(Continued)

| Reference | Title |
|---|---|
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Webs

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , and D .

(3) Dents are permitted between LBL 60.6 and RBL 60.6 only as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Detail E .

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 1.5 inch away from the edge of a hole, the edge of other damage, or the material edge.

(b) A maximum of 0.25 inch in diameter after they are drilled and filled with a 1/4 inch diameter or smaller rivet.

1) Use a 2117-T3 or 2177-T4 aluminum rivet.

2) Drill the hole to the applicable diameter as given in 51-40-05.

3) Install the rivet without sealant.

B. Stringers and Stiffeners

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , and F .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Fittings

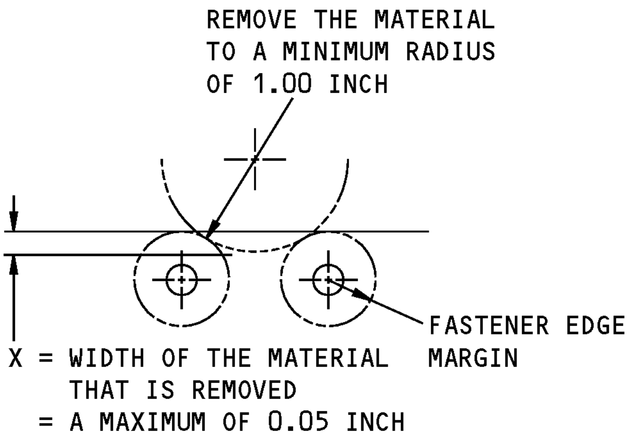
(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, C, and F.

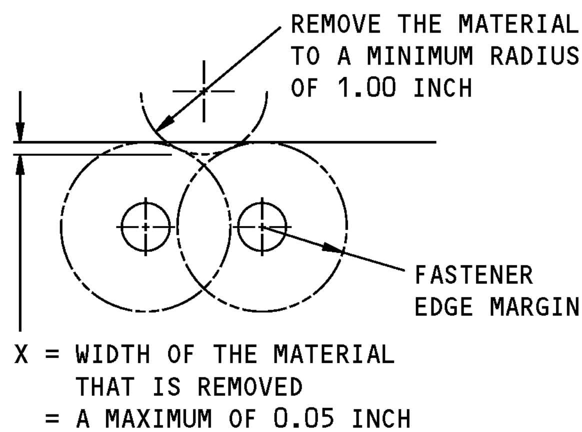
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



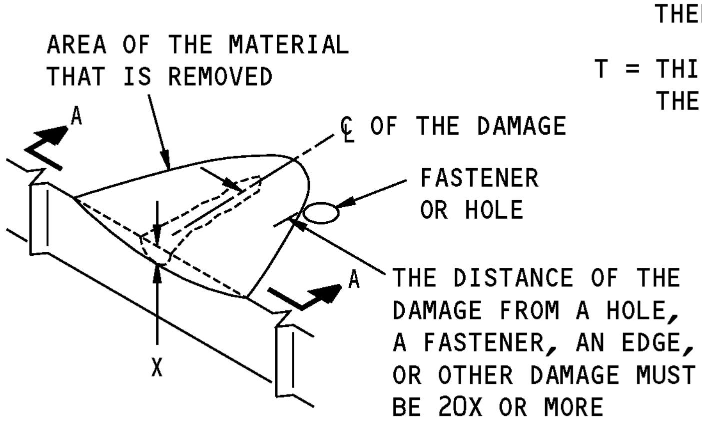

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A



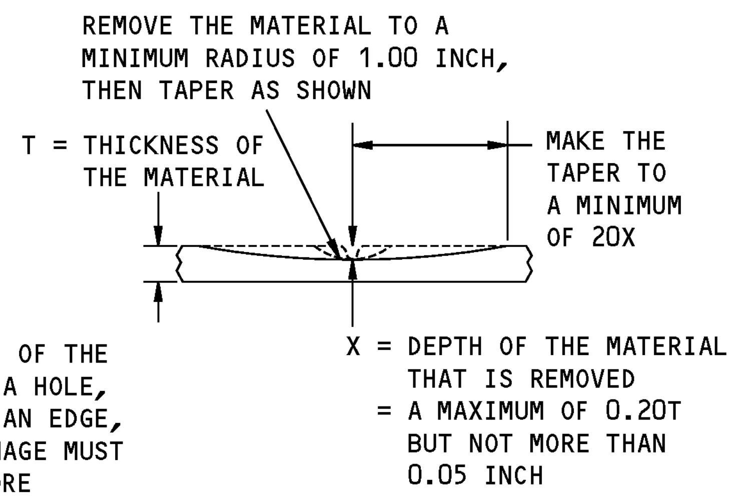

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B



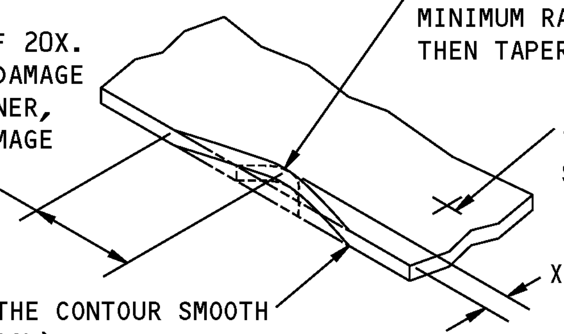

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C




A-A

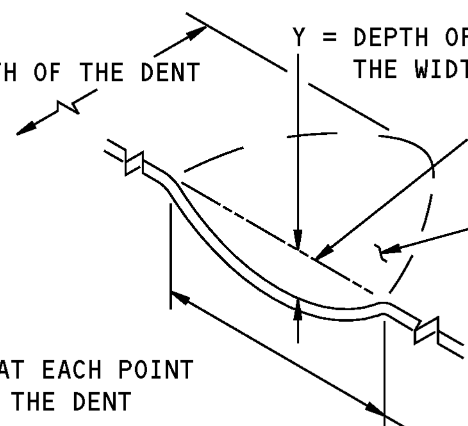
**Wing Center Section Spanwise Beams Allowable Damage Details**
**Figure 102 (Sheet 1 of 3)**




X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.05 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

D



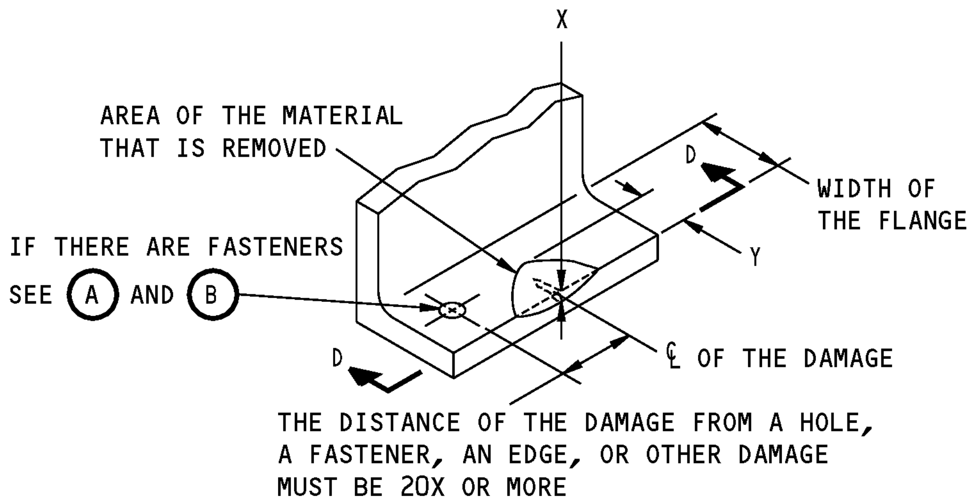

$\frac{W}{Y}$ MUST BE 30 OR MORE AT EACH POINT ALONG THE LENGTH OF THE DENT

Y = A MAXIMUM OF 0.10 INCH

DENT THAT IS PERMITTED

E

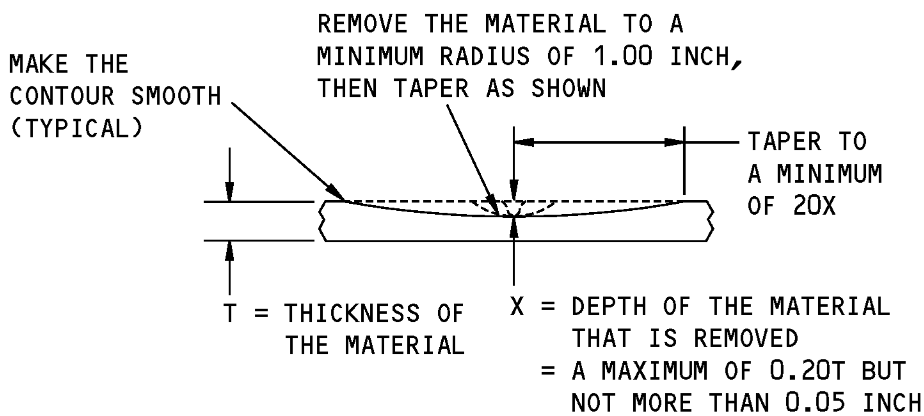
**Wing Center Section Spanwise Beams Allowable Damage Details**
**Figure 102 (Sheet 2 of 3)**




Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

F



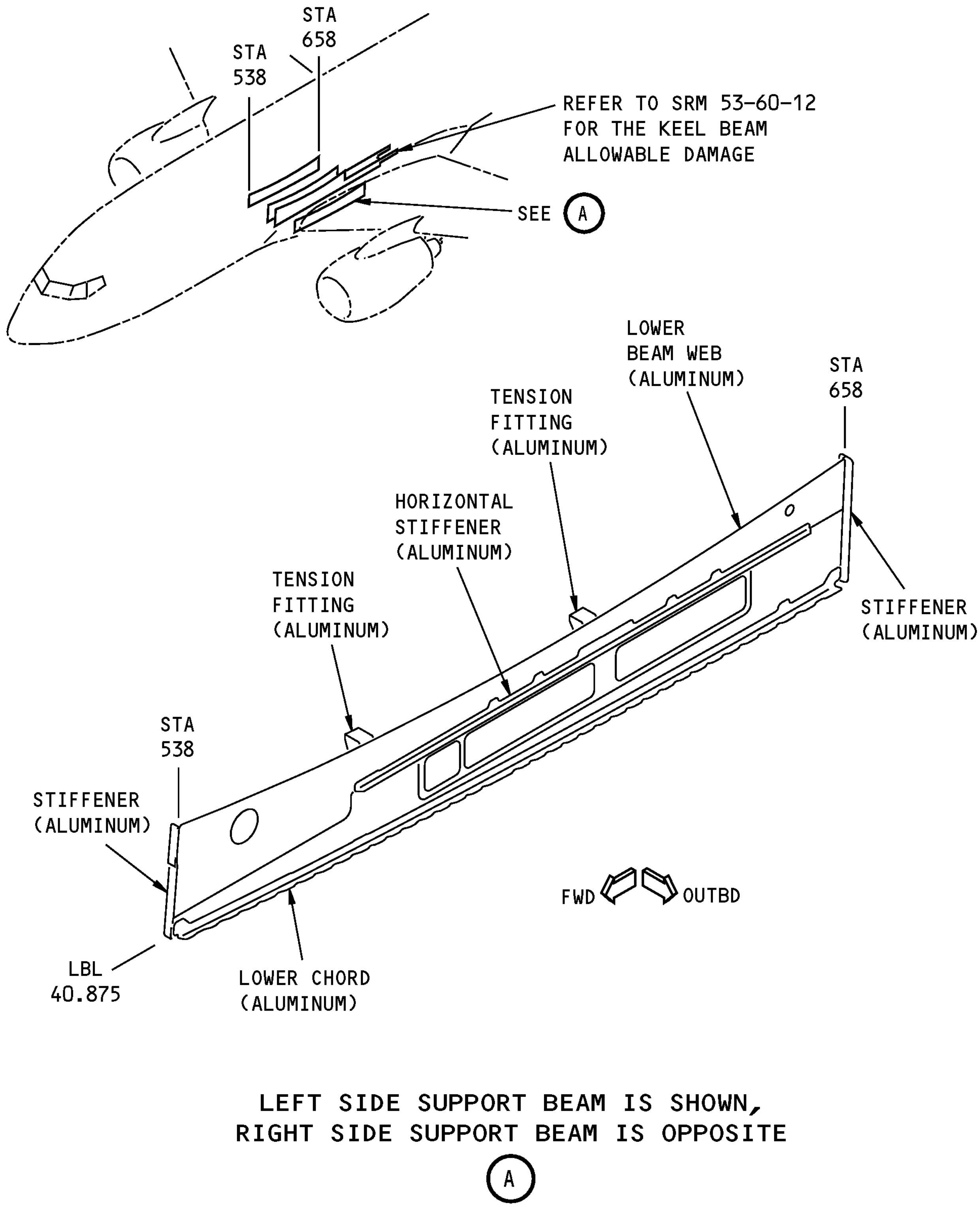

D-D

**Wing Center Section Spanwise Beams Allowable Damage Details**
**Figure 102 (Sheet 3 of 3)**

## ALLOWABLE DAMAGE 2 - WING CENTER SECTION SUPPORT BEAMS

### 1. Applicability

A. This subject is applicable to the wing center section support beams. Refer to Wing Center Section Support Beam Locations, Figure 101/ALLOWABLE DAMAGE 2.




**Wing Center Section Support Beam Locations**
**Figure 101**

## 2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damaged material as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasives and other materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the tools and equipment you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the areas that have rework. Refer to 51-20-01.

(3) Apply two layers of BMS 10-20, Type II primer to the areas that have rework. Refer to AMM 28-11-00/701.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-05 | FASTENER HOLE SIZES |
| AMM 51-21-00 | INTERIOR AND EXTERIOR FINISHES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Webs

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A, B, and F.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and F .

(3) Dents are permitted as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Detail G .

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 1.5 inch away from the edge of a hole, the edge of other damage, or the material edge

(b) A maximum of 0.25 inch in diameter after they are drilled and filled with a 1/4 inch diameter or smaller rivet.

1) Use a 2117-T3 or 2117-T4 aluminum rivet

2) Drill the hole to the applicable diameter as given in 51-40-05

3) Install the rivet without sealant.

B. Stiffeners

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A, B, and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and F .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Chord

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A, B, and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Fittings

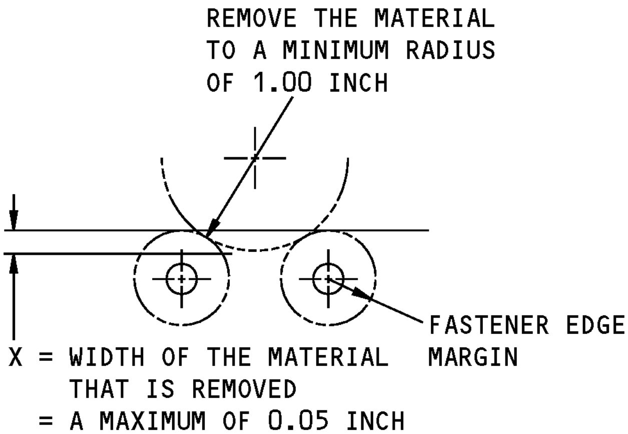
(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and D .

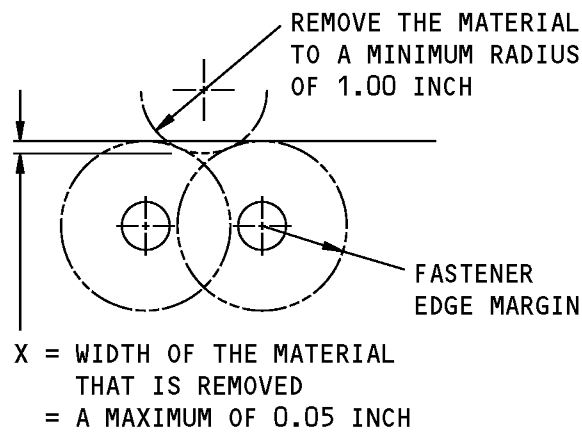
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



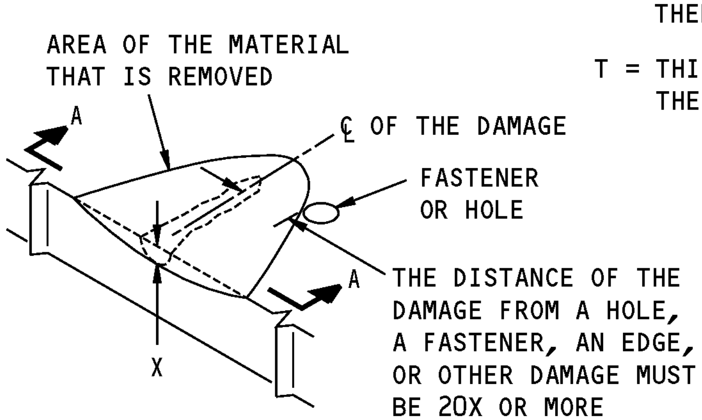

REMOVAL OF DAMAGED MATERIAL AT
EDGES WHERE THE FASTENER EDGE
MARGINS DO NOT HAVE AN OVERLAP

A



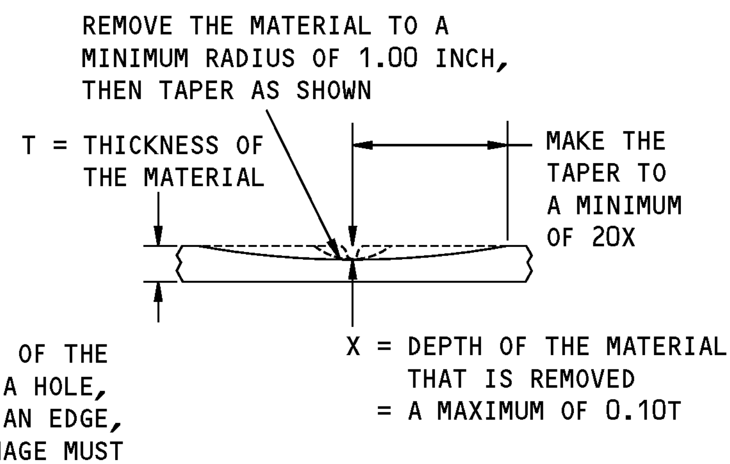

REMOVAL OF DAMAGED MATERIAL AT
EDGES WHERE THE FASTENER EDGE
MARGINS HAVE AN OVERLAP

B



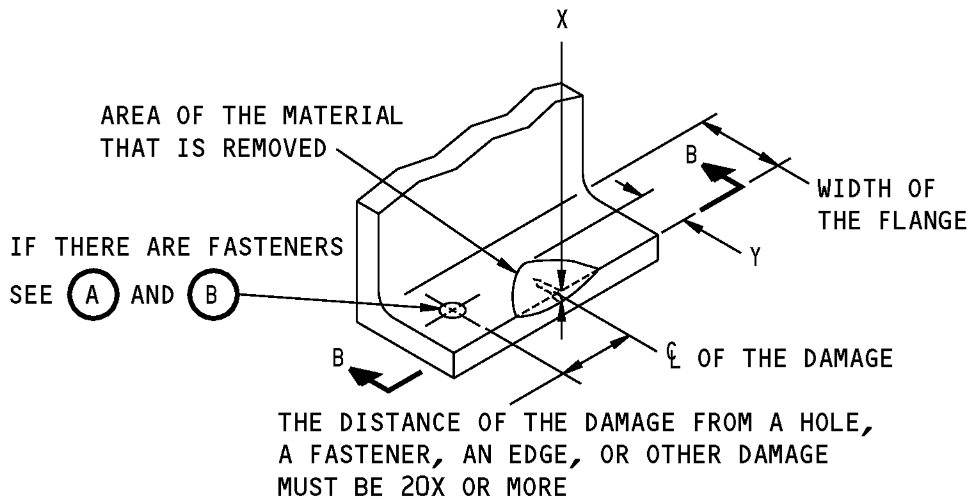

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

C



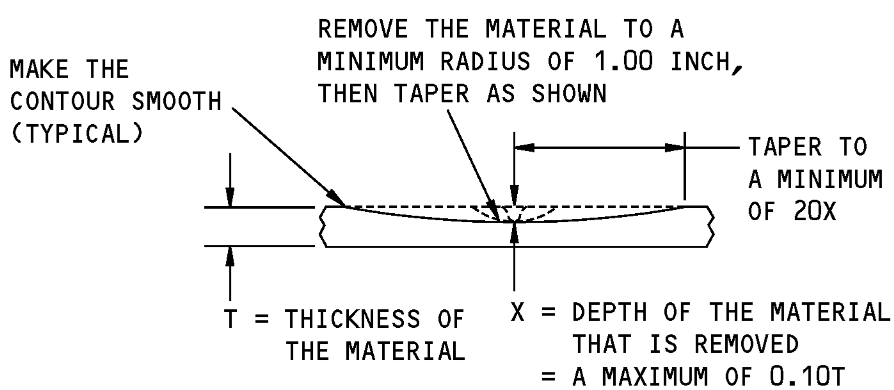

A-A

Wing Center Section Support Beam Allowable Damage
Figure 102 (Sheet 1 of 4)



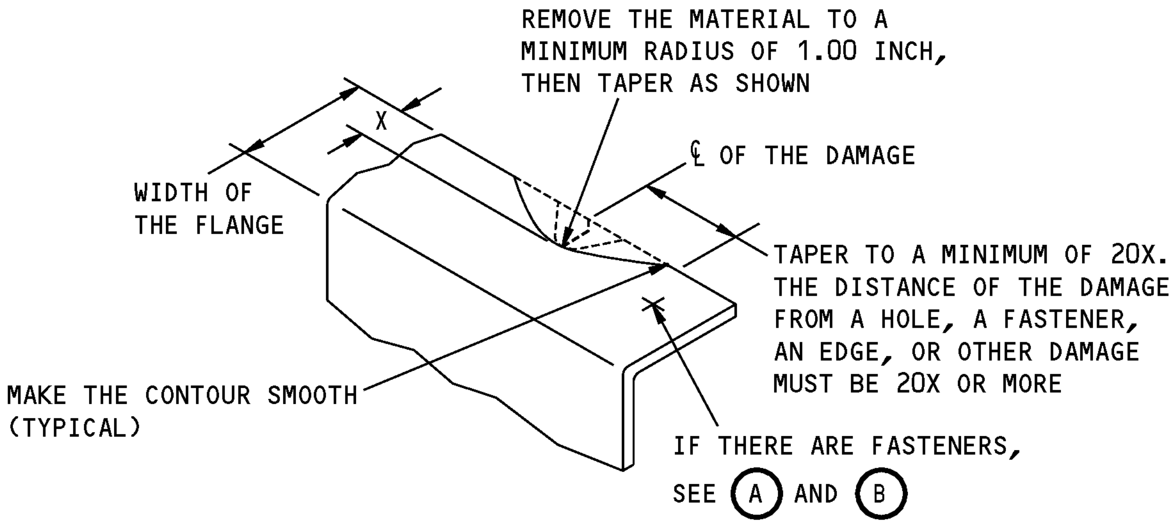

Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE
D




B-B

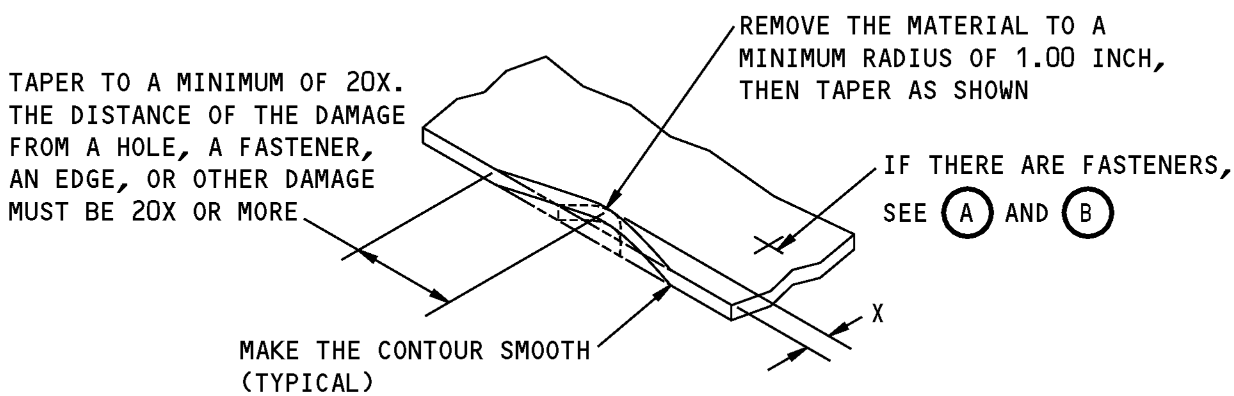
**Wing Center Section Support Beam Allowable Damage**
**Figure 102 (Sheet 2 of 4)**




X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE BUT NOT MORE THAN 0.05 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

E

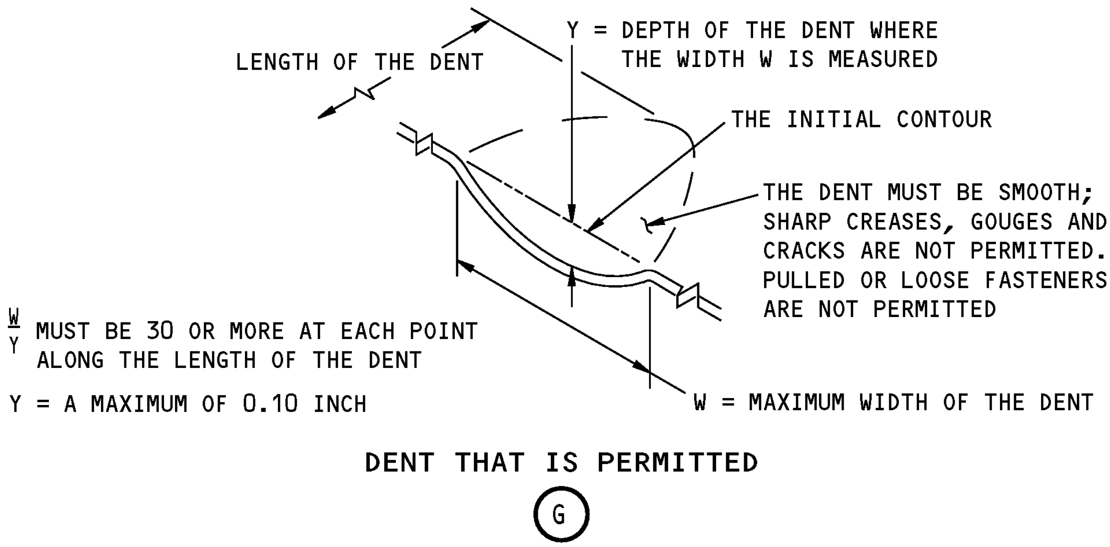



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.05 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

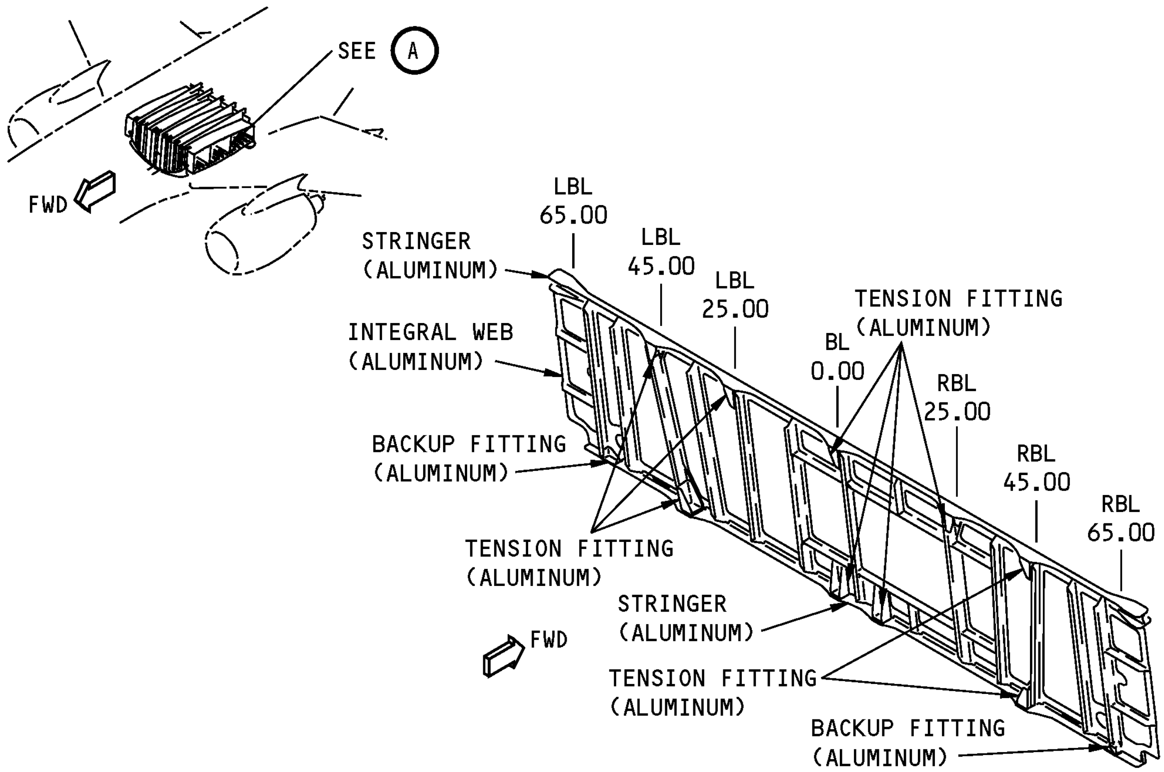
F

**Wing Center Section Support Beam Allowable Damage**
**Figure 102 (Sheet 3 of 4)**




**Wing Center Section Support Beam Allowable Damage**
**Figure 102 (Sheet 4 of 4)**

## REPAIR 1 - WING CENTER SECTION SPANWISE BEAMS



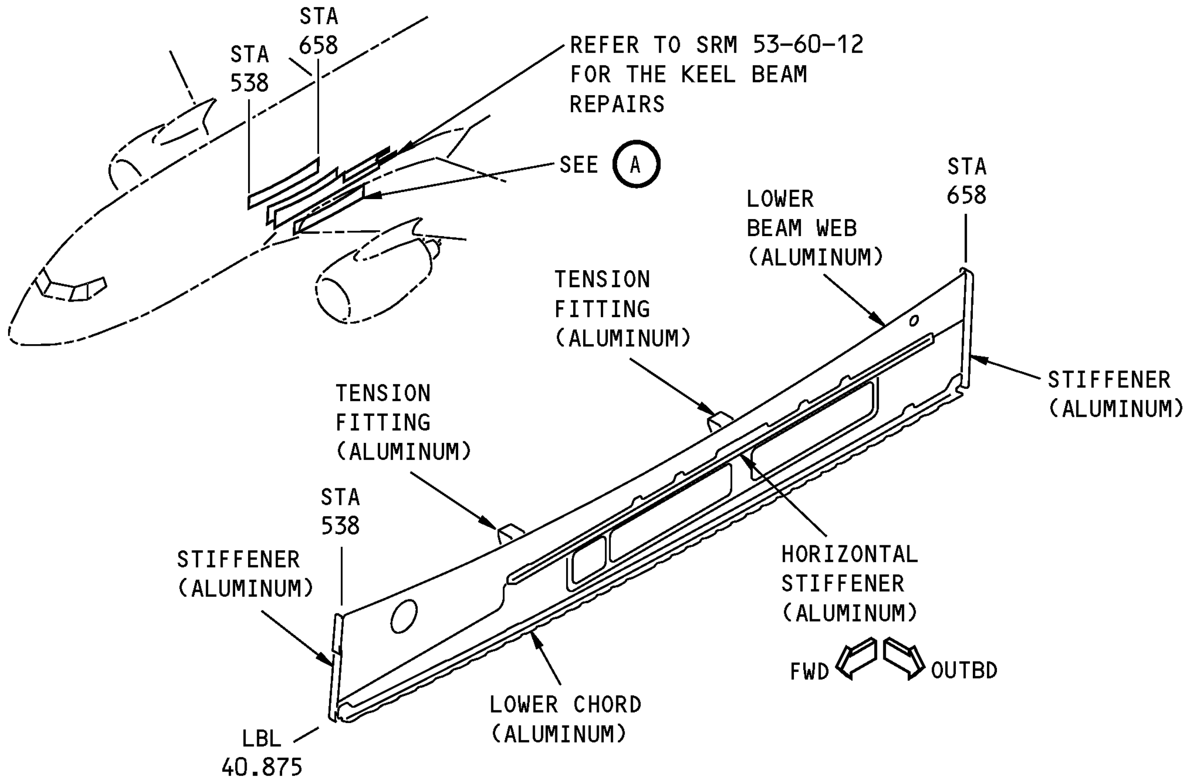

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

SPANWISE BEAM NUMBER 1 IS SHOWN,
SPANWISE BEAM NUMBER 2 IS ALMOST THE SAME

A

**Wing Center Section Spanwise Beam Repairs**
**Figure 201**

## REPAIR 2 - WING CENTER SECTION SUPPORT BEAMS




NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

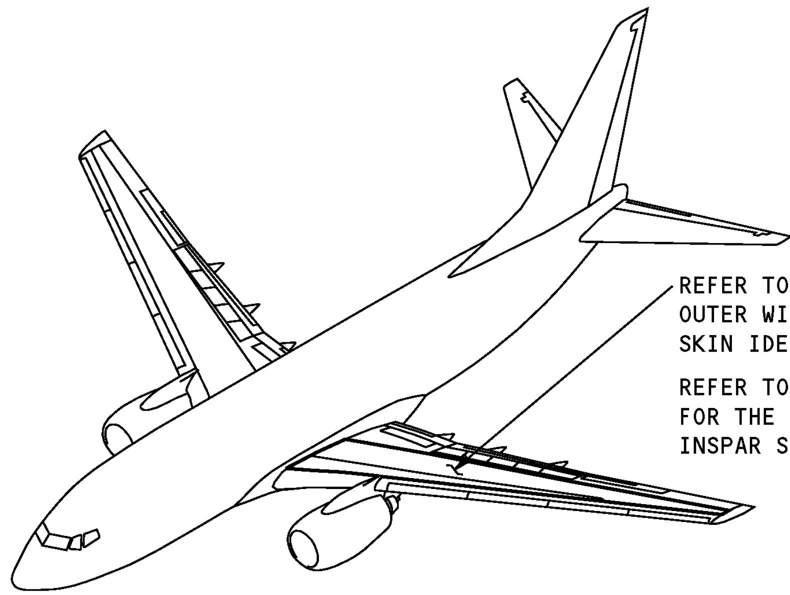
LEFT SIDE SUPPORT BEAM IS SHOWN,
RIGHT SIDE SUPPORT BEAM IS OPPOSITE

(A)

**Wing Center Section Support Beam Repairs**
**Figure 201**

## IDENTIFICATION 1 - OUTER WING UPPER INSPAR SKIN

### 1. Locator




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Outer Wing Upper Inspar Skin Location**
**Figure 1**

### 2. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 112A0001 | Functional Product Collector - Skin Panel |
| 112A0002 | Functional Product Collector - Skin Panel |
| 112A3001 | Panel Installation - Upper, Section 12, Wing |
| 112A3050 | Skin Splice Installation - Upper Forward, Wing |
| 112A3401 | Attach Strap Installation - Leading Edge to Wing, Upper |
| 112A3500 | Vortex Generator Installation - Upper Surface, Wing |

### 3. Applicability

A. Identification 1 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 1 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

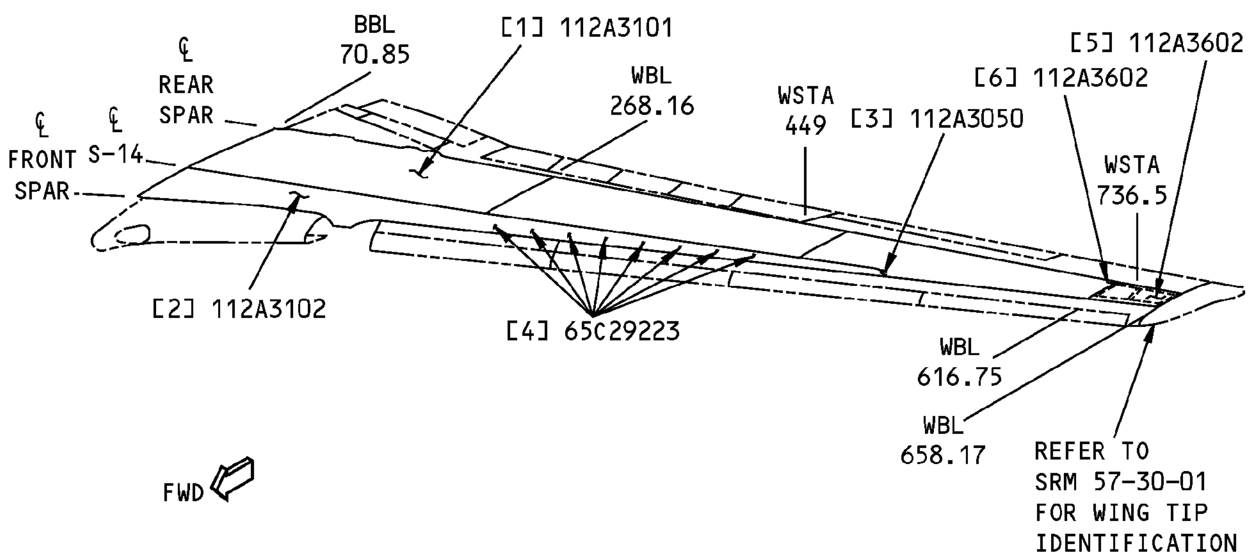
1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. List of Materials:

**Table 2**:

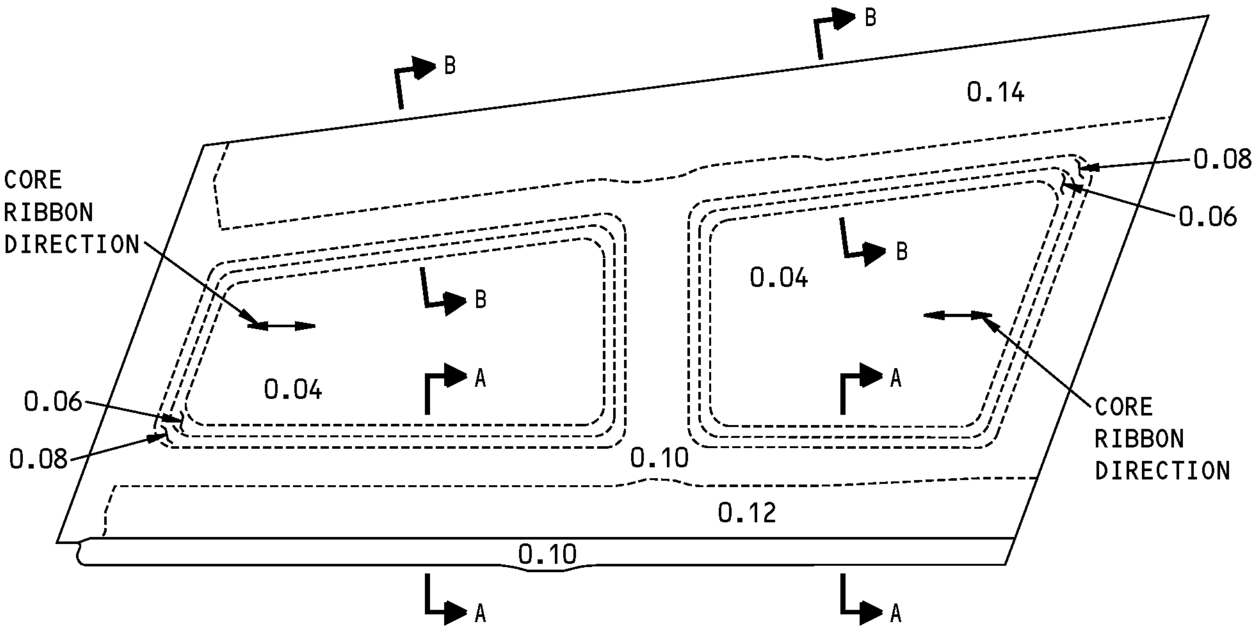
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Skin Panel Number 1 | | 7055-T7751 plate as given in BMS 7-307. Refer to the production drawings for the the machined thicknesses | |
| [2] | Skin Panel Number 2 | | 7055-T7751 plate as given in BMS 7-307. Refer to the production drawings for the machined thicknesses | |
| [3] | Splice Plate | 0.040 (1.02) | 6013-T4 sheet as given in AMS 43476 | |
| [4] | Vortex Generator | | AND10133-0602 2024-T3511 extrusion. (Optional: BAC1503-100093) | |
| [5] | Wing Tip Upper Skin Panel | 0.312 (7.937) | 7050-T7451 plate. Refer to Figure 3 for the chem-mill thicknesses | |
| [6] | Strap | 0.100 (2.54) | 7075-T6 sheet | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

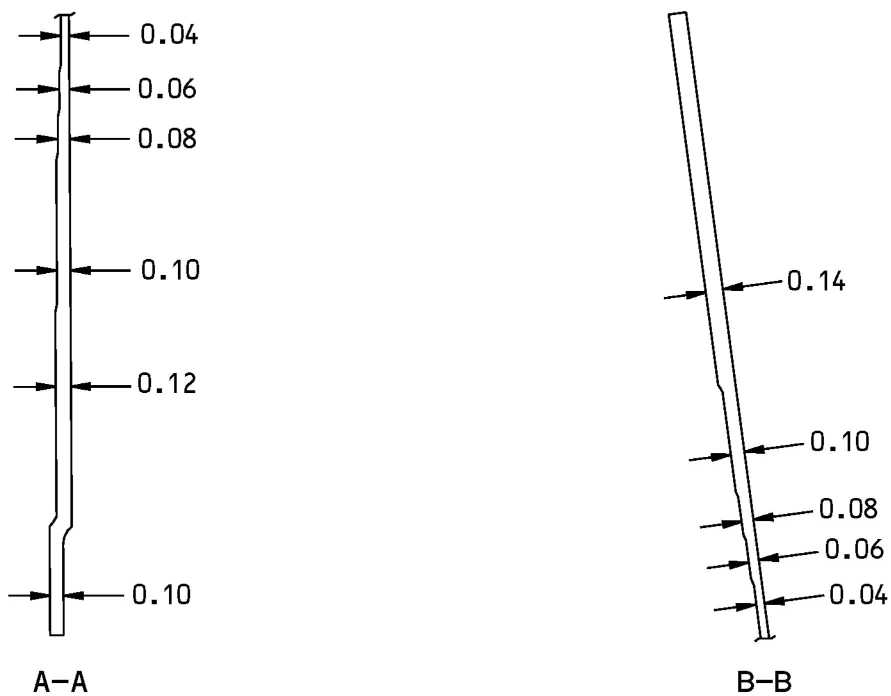



NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Outer Wing Upper Inspar Skin Identification**
**Figure 2**

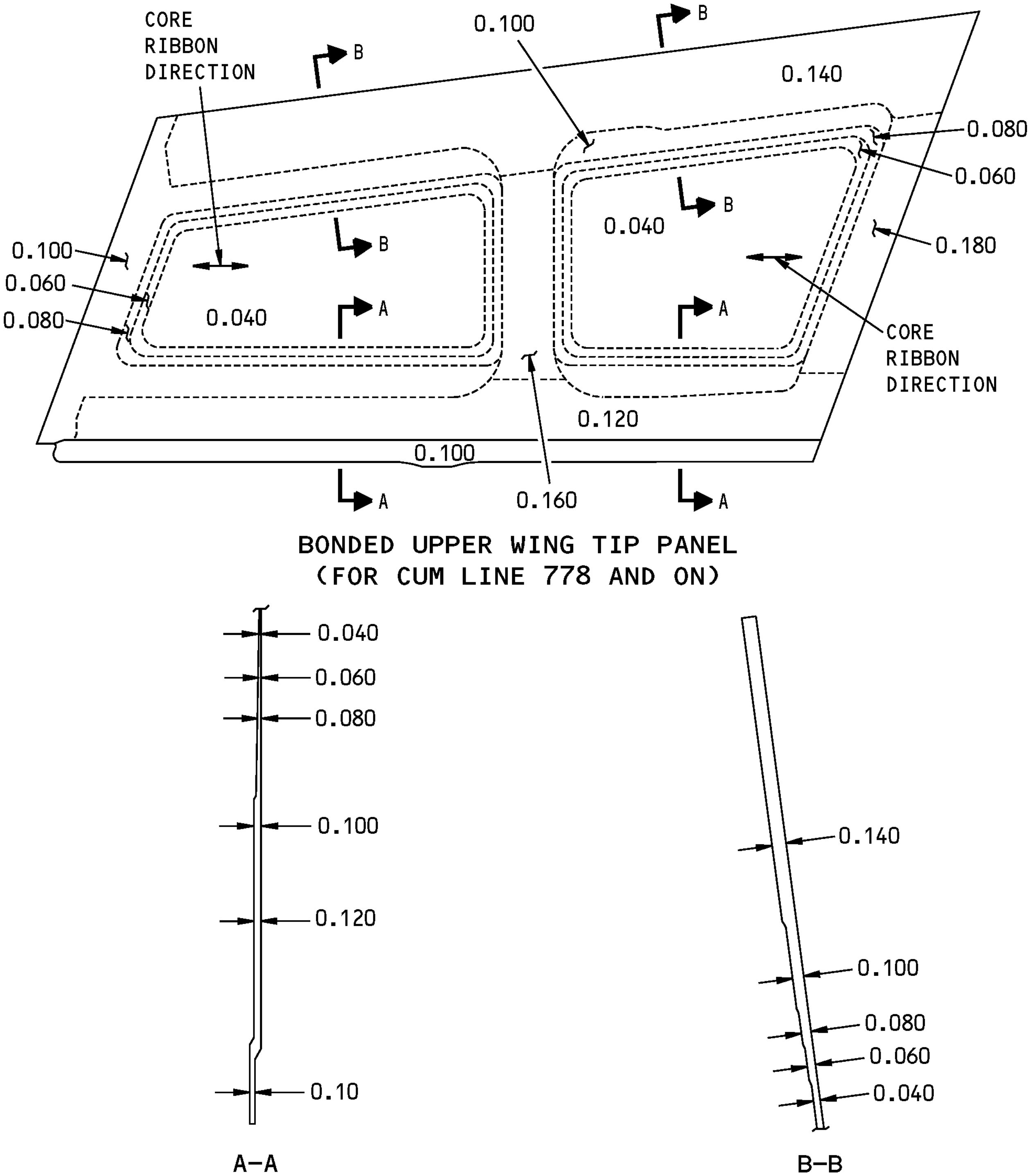



BONDED UPPER WING TIP PANEL
(FOR CUM LINE NUMBER 1 THRU 777)




NOTE: ALL DIMENSIONS ARE IN INCHES.

**Chem-mill Thicknesses for Figure 2, Item [5]**
**Figure 3 (Sheet 1 of 2)**

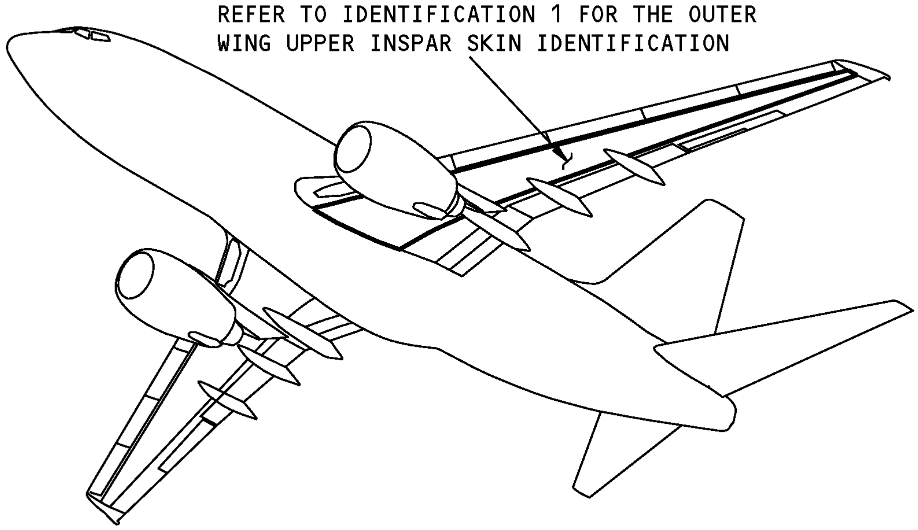



NOTE: ALL DIMENSIONS ARE IN INCHES.

**Chem-mill Thicknesses for Figure 2, Item [5]**
**Figure 3 (Sheet 2 of 2)**

## IDENTIFICATION 2 - OUTER WING LOWER INSPAR SKIN

### 1. Locator




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Outer Wing Lower Inspar Skin Location
Figure 1**

### 2. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 112A0001 | Functional Product Collector - Skin Panel |
| 112A0002 | Functional Product Collector - Skin Panel |
| 112A4001 | Panel Installation - Lower, Section 12, Wing |
| 112A4050 | Skin Splice Installation - Lower Aft, Wing |
| 112A4051 | Skin Splice Installation - Lower Forward, Wing |
| 112A4301 | Splice Installation - BBL 70.85, Lower Panel |
| 112A4601 | Panel Installation - Honeycomb, Lower, Wingtip, Wing |

### 3. Applicability

A. Identification 2 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

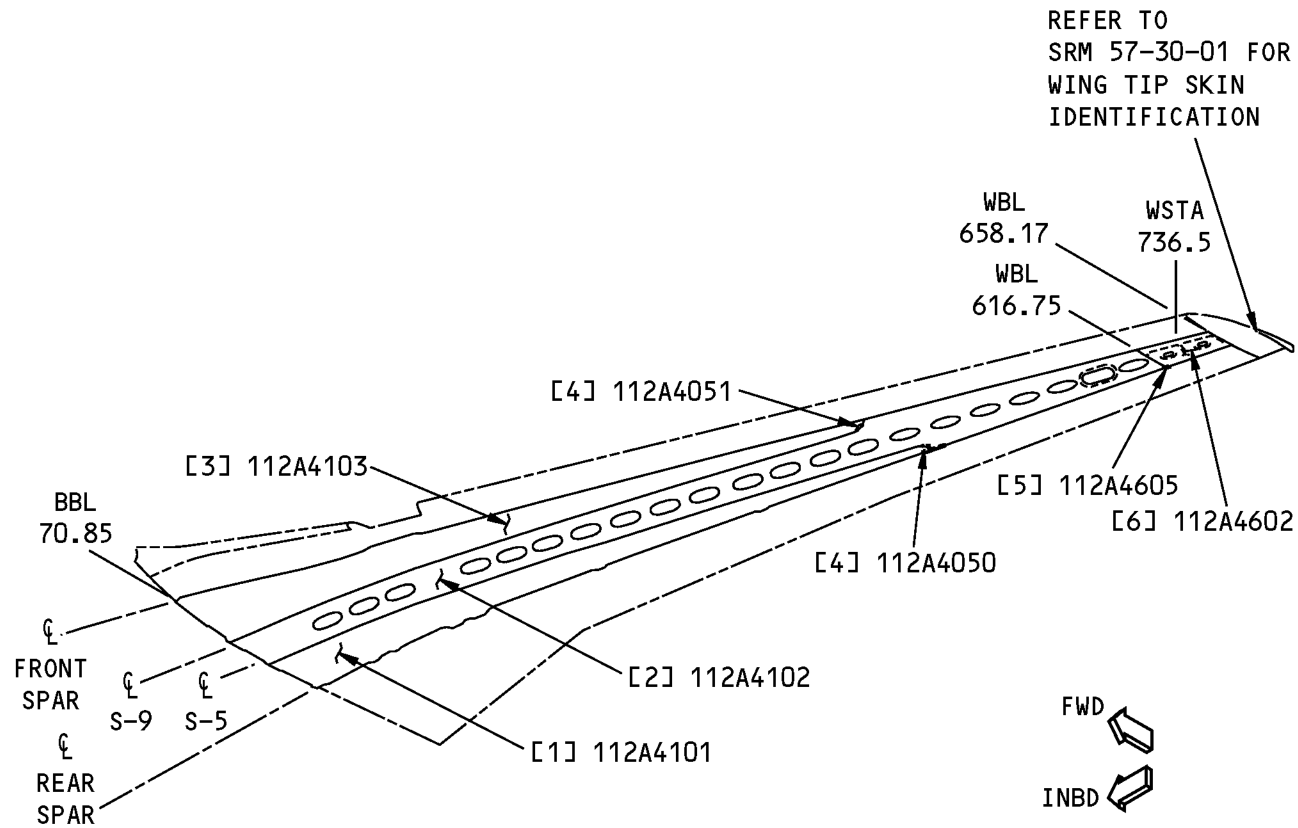
1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. List of Materials

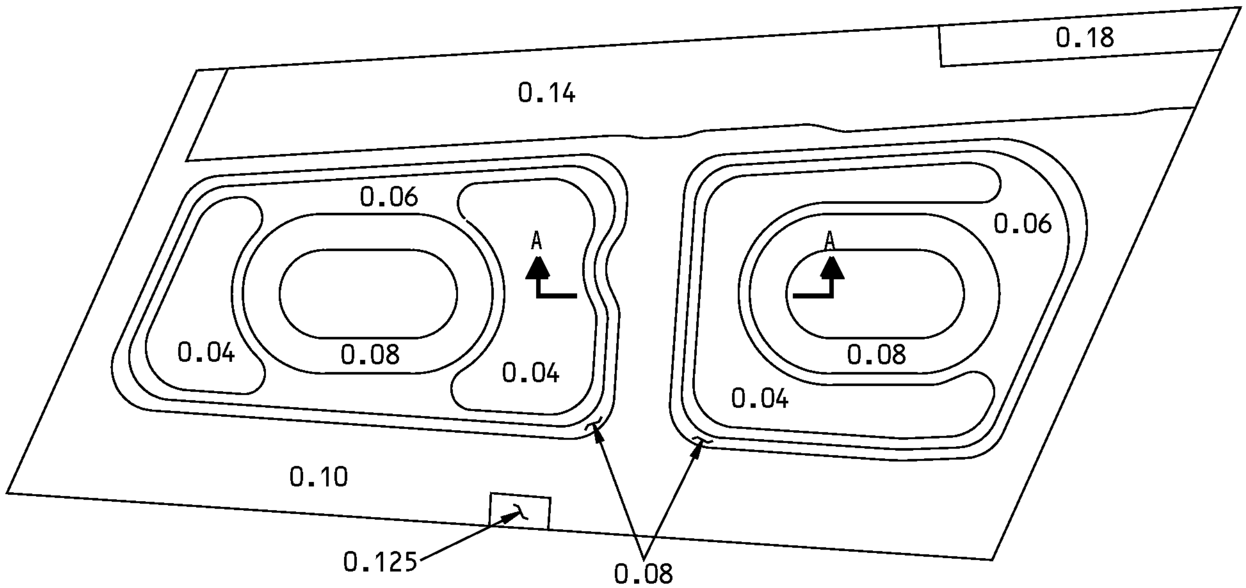
**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Skin Panel Number 1 | | 2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the machined thicknesses | |
| [2] | Skin Panel Number 2 | | 2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the machined thicknesses | |
| [3] | Skin Panel Number 3 | | 2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the machined thicknesses | |
| [4] | Splice Plate (2) | 0.040 (1.02) | 6013-T4 sheet as given in AMS 4347 | |
| [5] | Splice Plate | 0.100 (2.54) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [6] | Bonded Wing Tip Panel, Lower | | Bonded panel as given in BAC5514-5101 with 1 ply of BMS 5-101, Type II, Grade 10 adhesive. Refer to Figure 3 for the chem-mill thicknesses | |
| | Face Sheet, Inner | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Face Sheet, Inner | 0.032 (0.81) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Skin | 0.180 (4.57) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Core (2) | | 5052 aluminum honeycomb as given in BMS 4-4, Type 3-15NPA, Grade I | |

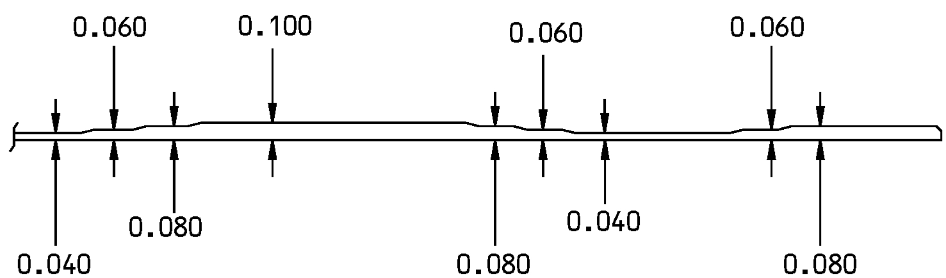
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Outer Wing Lower Inspar Skin Identification**
**Figure 2**



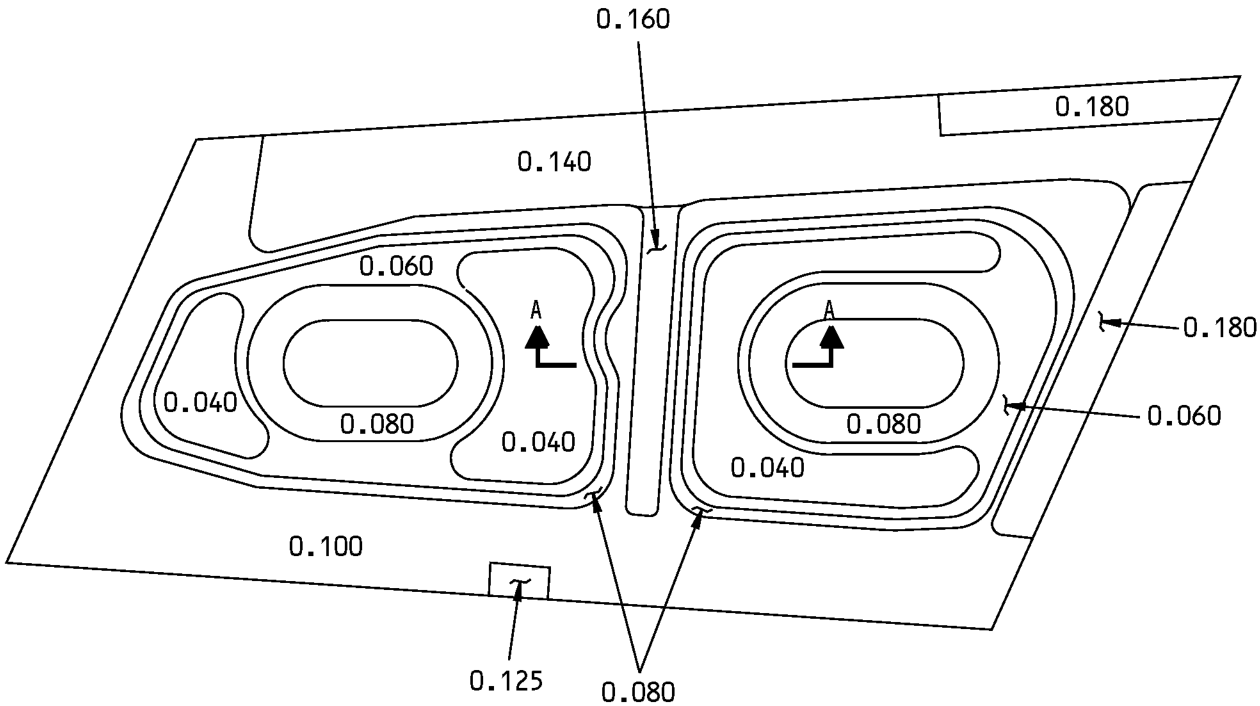

BONDED LOWER WING TIP PANEL
(FOR CUM LINE NUMBER 1 THRU 777)



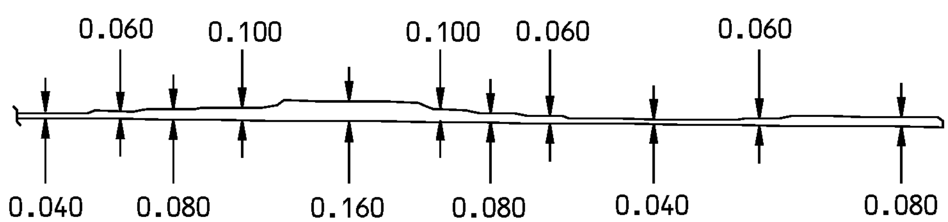

A-A

NOTE: ALL DIMENSIONS ARE IN INCHES.

**Chem-Mill Thicknesses for Figure 2, Item [6]**
**Figure 3 (Sheet 1 of 4)**



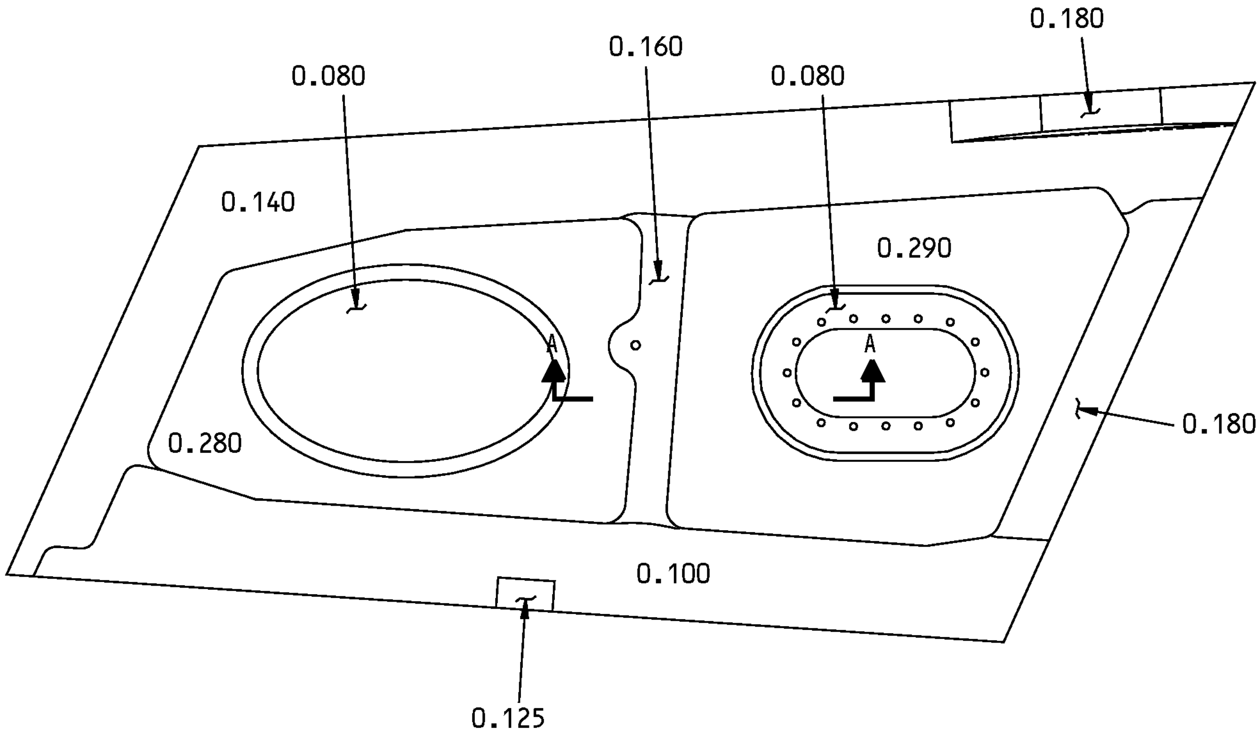

BONDED LOWER WING TIP PANEL
(FOR CUM LINE NUMBER 778 AND ON)



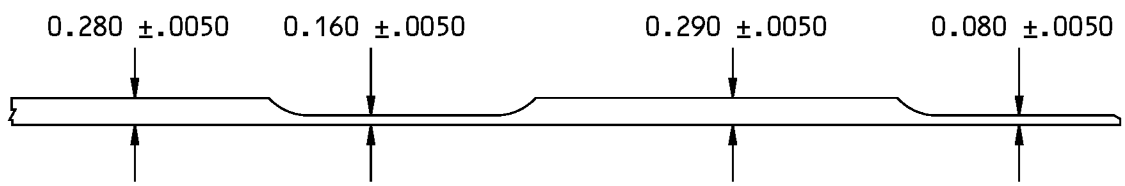

A-A

NOTE: ALL DIMENSIONS ARE THICKNESSES IN INCHES.

**Chem-Mill Thicknesses for Figure 2, Item [6]**
**Figure 3 (Sheet 2 of 4)**

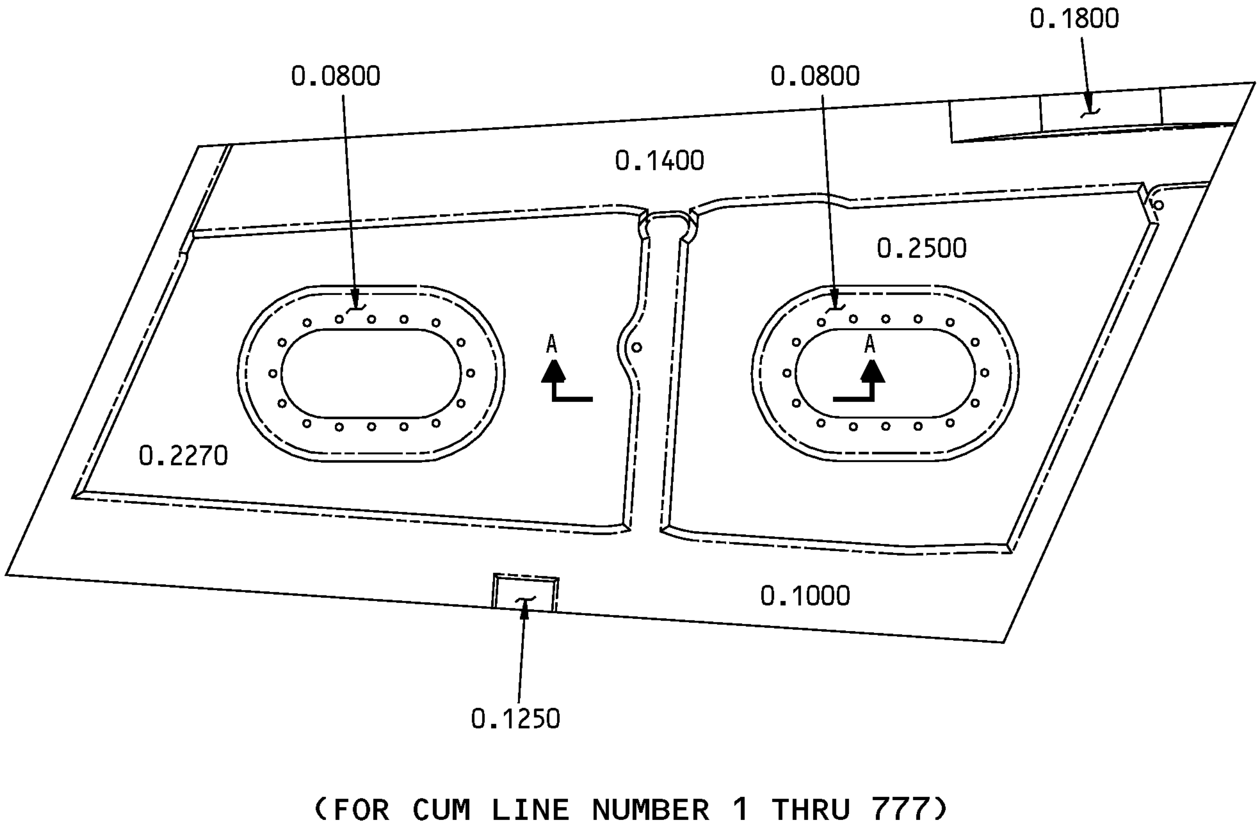


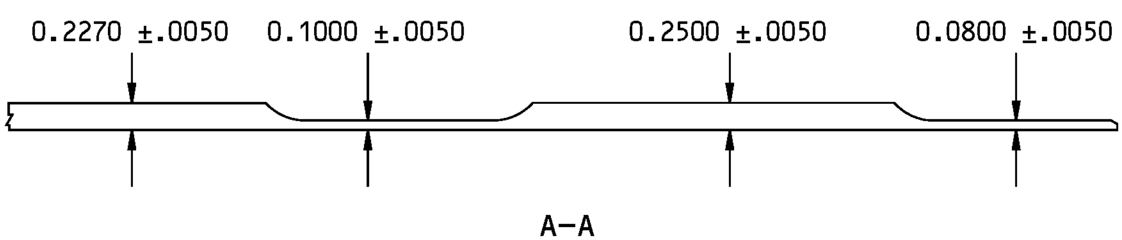

BONDED LOWER WING TIP PANEL
(FOR AIRPLANE CUM LINE NUMBER 789 AND ON WITH
FULL WINGLET PROVISION INSTALLATIONS)




A-A

**Chem-Mill Thicknesses for Figure 2, Item [6]**
**Figure 3 (Sheet 3 of 4)**




**Chem-Mill Thicknesses for Figure 2, Item [6]**
**Figure 3 (Sheet 4 of 4)**

## ALLOWABLE DAMAGE 1 - WING INSPAR SKIN INBOARD OF WBL 616.75

### 1. Applicability

A. This subject gives the allowable damage limits for the wing inspar skin inboard of the splice at WBL 616.75 shown in Wing Inspar Skin Location, Figure 101/ALLOWABLE DAMAGE 1.

B. Allowable Damage 1 is applicable to airplanes that have not had winglets installed.

C. Allowable Damage 1 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

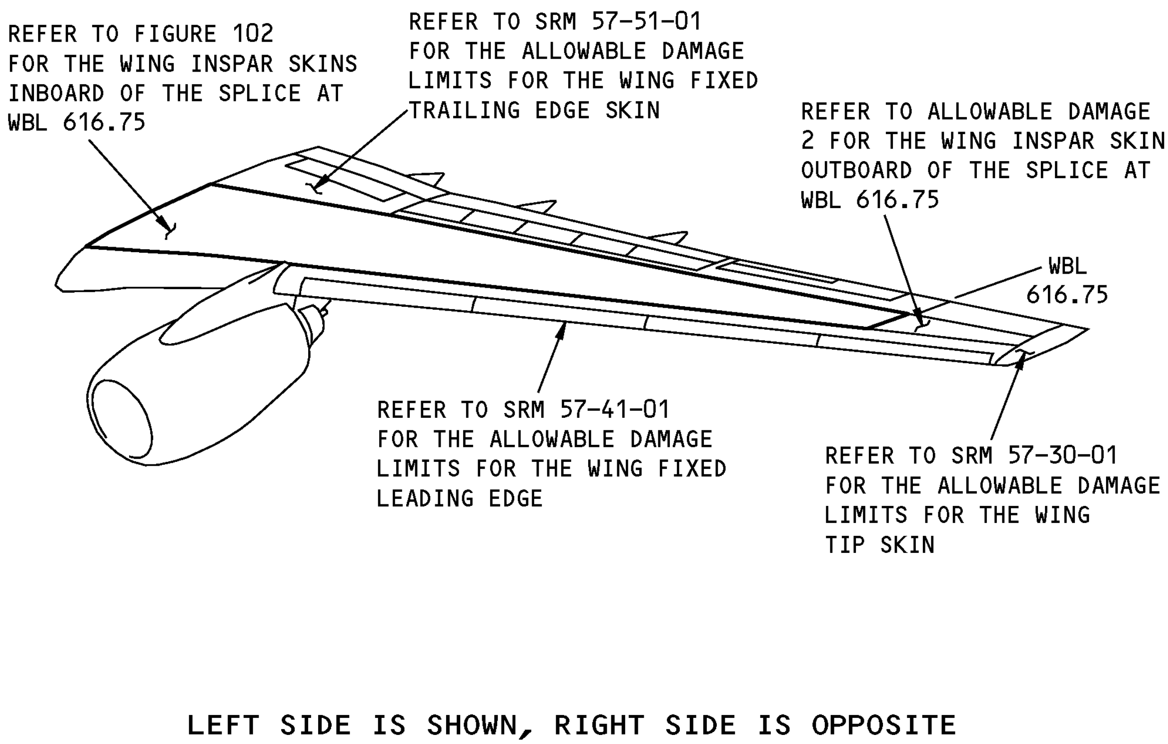
**NOTE**: Allowable Damage 1 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.




**Wing Inspar Skin Location**
**Figure 101**

2. **General**

A. Refer to Table 101 or Table 102 and Paragraph 4.A for the allowable damage limits for the upper inspar skin.

B. Refer to Table 103 or Table 104 and Paragraph 4.B for the allowable damage limits for the lower inspar skin.

C. Refer to Figure 103 (Sheet 1) for the definitions of skin areas.

D. If you have damage at a fuel tank door, do as follows before you remove the damage.

**WARNING:** IF YOU DO NOT SUPPLY GOOD AIRFLOW YOU WILL HAVE DANGEROUS AND EXPLOSIVE FUEL VAPORS IN THE WORK AREA WHICH CAN CAUSE INJURY TO PERSONS.

(1) Remove the fuel from the fuel tanks and make sure there is a good airflow in the damage area. Refer to AMM 28-11-00 and AMM 28-23-00 for the procedures.

(2) Remove the fuel tank access door. Refer to AMM 28-11-11.

E. Remove the damage as necessary. Refer to 51-10-02.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

F. After you remove the damage, do the procedures that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the skins of the lower inspar and fuel tank or vent door surrounds.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

(2) If you removed a fastener on the lower inspar skin that does not attach to the inspar rib, backup fitting, or nacelle support fitting, then do as follows:

(a) Cold Work the fastener hole with the High Interference Cold Work method. Refer to 51-40-09.

**NOTE**: Do not Cold Work the fastener hole on the lower inspar skin at an inspar rib, backup fitting, or nacelle support fitting. Refer to 51-40-02 for fastener installation procedures.

(3) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(4) Apply the finishes to the surfaces of the reworked areas as follows.

**NOTE**: Make sure the aerodynamic smoothness on the external surfaces is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

**WARNING:** DO NOT APPLY A FINISH TO THE ELECTRICAL BOND SURFACE OF THE SKIN AT THE FUEL TANK ACCESS OR VENT DOOR SURROUND. IF THERE IS PRIMER ON THE ELECTRICAL BOND SURFACE OF THE DOOR, YOU MUST REMOVE THE PRIMER. THE ELECTRICAL BOND IS NECESSARY TO PREVENT POSSIBLE LIGHTNING DAMAGE AT THE FUEL TANK.

(a) Apply one layer of BMS 10-79, Type III primer. Refer to SOPM 20-44-04.

(b) Apply one layer of BMS 10-60, Type II enamel to the external skin surface. Refer to AMM 51-21-00/701.

(c) Apply one layer of BMS 10-20, Type II primer to the internal skin surface. Refer to AMM 51-21-00/701.

Table 101:

| ALLOWABLE DAMAGE LIMITS FOR THE UPPER SKIN FOR AIRPLANES THAT DO NOT HAVE WINGLETS | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| ZONE (WING STATION) | MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR | | | | | | | | | | TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO REAR SPAR (SQ. IN.) |
| | REAR SPAR | | STIFFENER NO. 1 THROUGH STIFFENER NO. 6 | | STIFFENER NO. 7 THROUGH STIFFENER NO. 11 | | STIFFENER NO. 12 THROUGH STIFFENER NO. 21 | | FRONT SPAR | | |
| | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | |
| WBL91-WS204 | 0.02 | 0.04 | 0.02 | 0.02 | 0.02 | 0.02 | 0.00 | 0.00 | 0.02 | 0.02 | 0.15 |
| WS204-WS253 | 0.01 | 0.08 | 0.00 | 0.00 | 0.00 | 0.00 | 0.02 | 0.02 | 0.04 | 0.02 | 0.10 |
| WS253-WS326 | 0.01 | 0.07 | 0.02 | 0.02 | 0.02 | 0.02 | 0.01 | 0.02 | 0.03 | 0.02 | 0.20 |
| WS326-WS432 | 0.01 | 0.06 | 0.01 | 0.02 | 0.00 | 0.00 | 0.01 | 0.01 | 0.03 | 0.02 | 0.10 |
| WS432-WS537 | 0.01 | 0.05 | 0.00 | 0.00 | 0.00 | 0.00 | 0.00 | 0.00 | 0.02 | 0.02 | 0.05 |
| WS537-WS617 | 0.01 | 0.03 | — | – | 0.01 | 0.01 | 0.00 | 0.00 | 0.02 | 0.02 | 0.05 |
| WS617-WBL616.7 | 0.01 | 0.02 | — | – | 0.01 | 0.02 | 0.01 | 0.02 | 0.01 | 0.02 | 0.05 |

Table 102:

| ALLOWABLE DAMAGE LIMITS FOR THE UPPER SKIN FOR AIRPLANES THAT HAVE WINGLETS | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| ZONE (WING STATION) | MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR | | | | | | | | | | TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO REAR SPAR (SQ. IN.) |
| | REAR SPAR | | STIFFENER NO. 1 THROUGH STIFFENER NO. 6 | | STIFFENER NO. 7 THROUGH STIFFENER NO. 11 | | STIFFENER NO. 12 THROUGH STIFFENER NO. 21 | | FRONT SPAR | | |
| | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | |
| WBL91-WS204 | 0.02 | 0.04 | 0.01 | 0.01 | 0.01 | 0.02 | 0.00 | 0.00 | 0.02 | 0.02 | 0.15 |
| WS204-WS253 | 0.01 | 0.08 | 0.00 | 0.00 | 0.00 | 0.00 | 0.02 | 0.02 | 0.04 | 0.02 | 0.10 |
| WS253-WS326 | 0.01 | 0.07 | 0.00 | 0.00 | 0.02 | 0.02 | 0.01 | 0.02 | 0.03 | 0.02 | 0.20 |
| WS326-WS432 | 0.01 | 0.06 | 0.01 | 0.02 | 0.00 | 0.00 | 0.00 | 0.00 | 0.03 | 0.02 | 0.10 |

| ALLOWABLE DAMAGE LIMITS FOR THE UPPER SKIN FOR AIRPLANES THAT HAVE WINGLETS | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| ZONE (WING STATION) | MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR | | | | | | | | | TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO REAR SPAR (SQ. IN.) |
| | REAR SPAR | | STIFFENER NO. 1 THROUGH STIFFENER NO. 6 | | STIFFENER NO. 7 THROUGH STIFFENER NO. 11 | | STIFFENER NO. 12 THROUGH STIFFENER NO. 21 | | FRONT SPAR | | |
| | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | |
| WS432-WS537 | 0.01 | 0.05 | 0.00 | 0.00 | 0.00 | 0.00 | 0.00 | 0.00 | 0.02 | 0.02 | 0.05 |
| WS537-WS617 | 0.01 | 0.03 | — | – | 0.00 | 0.00 | 0.00 | 0.00 | 0.02 | 0.02 | 0.05 |
| WS617-WBL616.7 | 0.01 | 0.02 | — | – | 0.01 | 0.02 | 0.01 | 0.02 | 0.01 | 0.02 | 0.05 |

**Table 103**:

| ALLOWABLE DAMAGE LIMITS FOR THE LOWER SKIN FOR AIRPLANES THAT DO NOT HAVE WINGLETS | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|
| ZONE (WING STATION) | MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR | | | | | | | | TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO) REAR SPAR (SQ. IN.) |
| | REAR SPAR | | STIFFENER NO. 1 THROUGH STIFFENER NO. 6 | | STIFFENER NO. 8 THROUGH STIFFENER NO. 14 | | FRONT SPAR | | |
| | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | |
| WBL71-WS204 | 0.03 | 0.06 | 0.02 | 0.04 | 0.02 | 0.07 | 0.02 | 0.03 | 0.40 |
| WS204-WS253 | 0.05 | 0.10 | 0.03 | 0.10 | 0.03 | 0.10 | 0.06 | 0.09 | 0.40 |
| WS253-WS326 | 0.04 | 0.08 | 0.03 | 0.02 | 0.03 | 0.10 | 0.04 | 0.06 | 0.30 |
| WS326-WS432 | 0.03 | 0.06 | 0.02 | 0.03 | 0.03 | 0.03 | 0.02 | 0.03 | 0.10 |
| WS432-WS537 | 0.02 | 0.05 | 0.02 | 0.05 | 0.00 | 0.00 | 0.00 | 0.00 | 0.05 |
| WS537-WS617 | 0.01 | 0.03 | 0.01 | 0.03 | 0 .00 | 0.00 | 0.00 | 0.00 | 0.05 |
| WS617-WBL616.7 | 0.01 | 0.02 | 0.01 | 0.02 | 0.01 | 0.02 | 0.01 | 0.02 | 0.05 |

Table 104:

| ALLOWABLE DAMAGE LIMITS FOR THE LOWER SKIN FOR AIRPLANES THAT HAVE WINGLETS | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|
| ZONE (WING STATION) | MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR | | | | | | | | TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO) REAR SPAR (SQ. IN.) |
| | REAR SPAR | | STIFFENER NO. 1 THROUGH STIFFENER NO. 6 | | STIFFENER NO. 8 THROUGH STIFFENER NO. 14 | | FRONT SPAR | | |
| | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | MAX DEPTH (INCH) | MAX AREA LOSS (SQ. IN.) | |
| WBL71-WS204 | 0.03 | 0.06 | 0.02 | 0.04 | 0.01 | 0.07 | 0.02 | 0.03 | 0.40 |
| WS204-WS253 | 0.05 | 0.10 | 0.03 | 0.10 | 0.03 | 0.10 | 0.06 | 0.09 | 0.40 |
| WS253-WS326 | 0.04 | 0.08 | 0.03 | 0.02 | 0.03 | 0.10 | 0.04 | 0.06 | 0.30 |
| WS326-WS432 | 0.03 | 0.06 | 0.02 | 0.03 | 0.03 | 0.03 | 0.02 | 0.03 | 0.10 |
| WS432-WS537 | 0.02 | 0.05 | 0.02 | 0.05 | 0.00 | 0.00 | 0.00 | 0.00 | 0.05 |
| WS537-WS617 | 0.01 | 0.03 | 0.01 | 0.03 | 0 .00 | 0.00 | 0.00 | 0.00 | 0.05 |
| WS617-WBL616.7 | 0.01 | 0.02 | 0.00 | 0.00 | 0.00 | 0.00 | 0.01 | 0.02 | 0.02 |

Table 105:

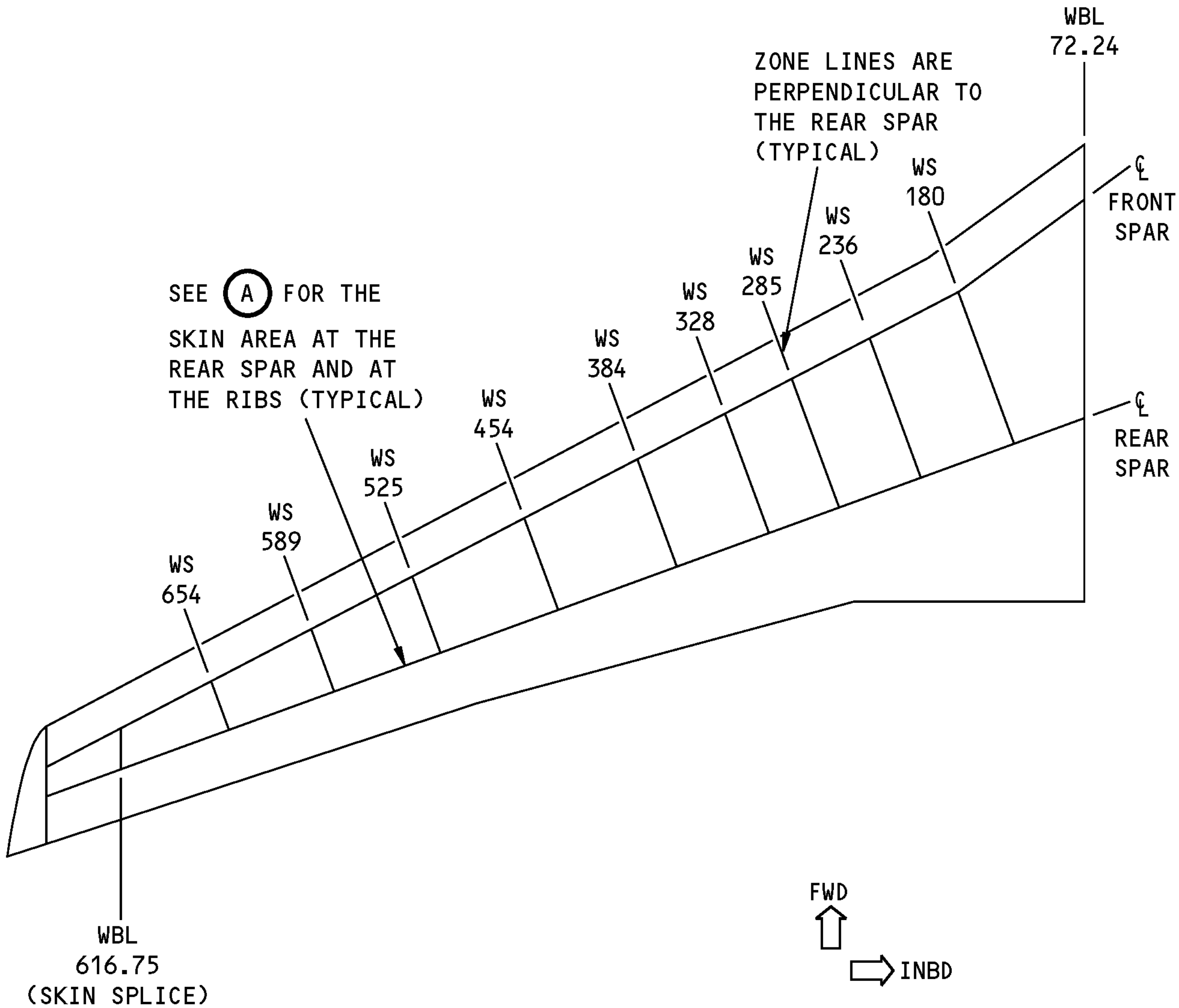
| INITIAL FASTENER | MAXIMUM DEPTH OF MATERIAL REMOVAL FROM A FASTENER HEAD (INCH) | | | |
|---|---|---|---|---|
| | 3/16 DIA | 1/4 DIA | 4/16 DIA | 3/8 DIA |
| ALUMINUM RIVET WITH 82°/30° COUNTERSINK | 0.01 | 0.01 | 0.01 | 0.01 |
| BACB30VU()K() BACB30NW()K() | 0.00 | 0.00 | 0.00 | 0.00 |
| BACB30PT()K() | 0.00 | 0.00 | 0.00 | 0.00 |

Table 106:

| INITIAL FASTENER | FASTENER | MINIMUM THICKNESS OF THE SKIN THAT REMAINS AFTER CLEANUP | REPLACEMENT FASTENER |
|---|---|---|---|
| ALUMINUM RIVET WITH 82°/30° COUNTERSINK | 3/16 | 0.12 | BACB30NY6K()Y BOLT WITH A BACC30AC6 COLLAR |
| | 1/4 | 0.20 | BACB30NY8K()Y BOLT WITH A BACC30AC8 COLLAR |
| | 5/16 | 0.20 | BACB30NY10K()Y BOLT WITH A BACC30AC10 COLLAR |
| | 3/8 | 0.24 | BACB30NY12K()Y BOLT WITH A BACB30AC12 COLLAR |

| INITIAL FASTENER | FASTENER | MINIMUM THICKNESS OF THE SKIN THAT REMAINS AFTER CLEANUP | REPLACEMENT FASTENER |
|---|---|---|---|
| BACB30VU()K() BOLT WITH A BACC30BP COLLAR<br>BACB30NW()K() BOLT WITH A BACC30M COLLAR | 3/16 | 0.08 | BACB30NW6K()X BOLT WITH A BACC30M6 COLLAR |
| | 1/4 | 0.09 | BACB30NW8K()X BOLT WITH A BACC30M8 COLLAR |
| | 5/16 | 0.10 | BACB30NW10K()X BOLT WITH A BACC30M10 COLLAR |
| | 3/8 | 0.12 | BACB30NW12K()X BOLT WITH A BACC30M12 COLLAR |
| BACB30PT()K() BOLT WITH A BACN10WM NUT | 3/16 | 0.08 | BACB30NW6K()X BOLT WITH A BACC30AC6 COLLAR |
| | 1/4 | 0.09 | BACB30NW8K()X BOLT WITH A BACC30AC8 COLLAR |
| | 5/16 | 0.10 | BACB30NW10K()X BOLT WITH A BACC30AC10 COLLAR |
| | 3/8 | 0.12 | BACB30NW12K()X BOLT WITH A BACC30AC12 COLLAR |

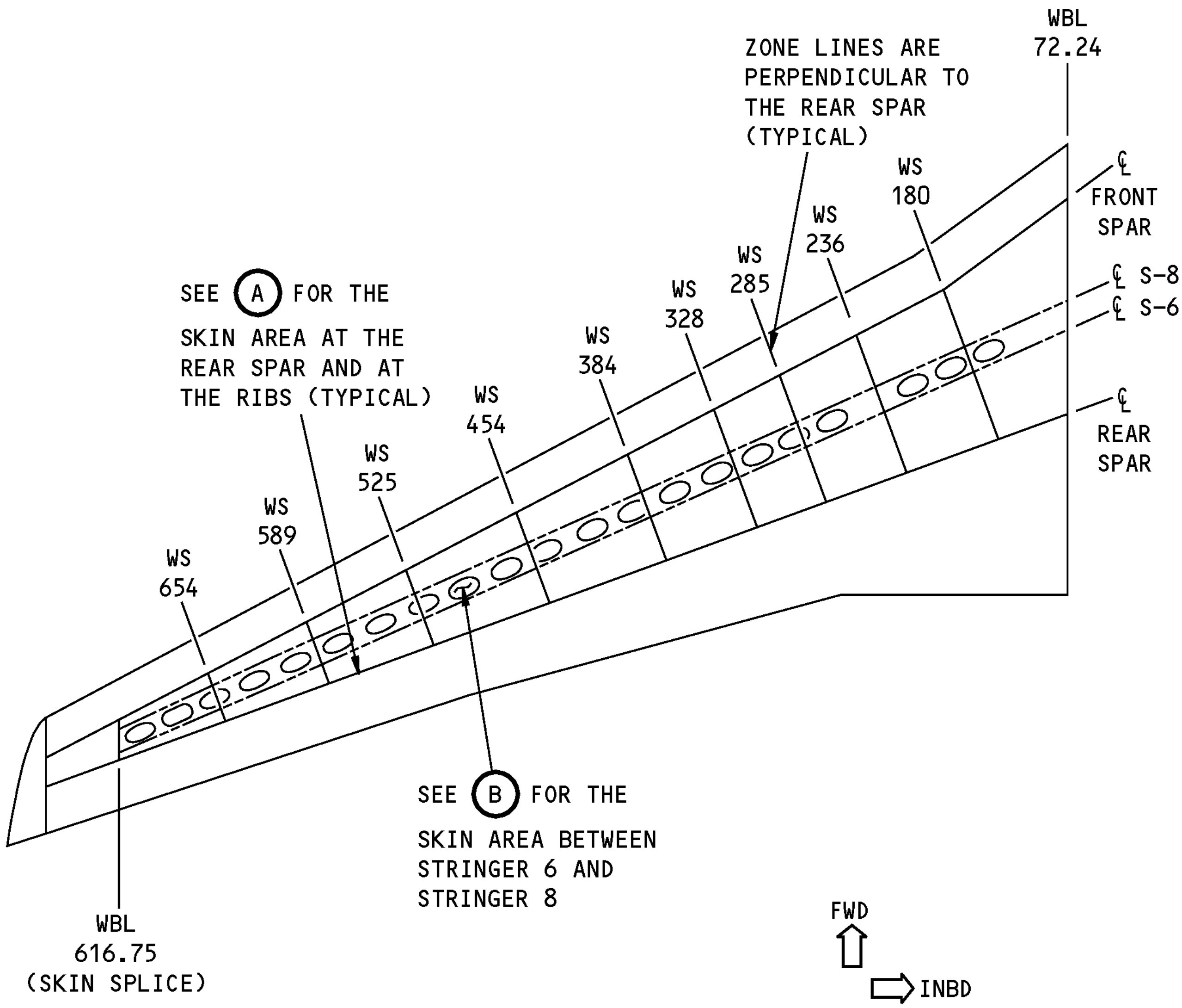
**NOTE**: Refer to Table 107 if the minimum thickness of the skin that remains is less than is given for an aluminum rivet with a 82°/30°countersink.




LEFT SIDE UPPER SKIN IS SHOWN,
RIGHT SIDE IS OPPOSITE

NOTE: THE UPPER INSPAR WING SKIN IS MADE OF MACHINED ALUMINUM.

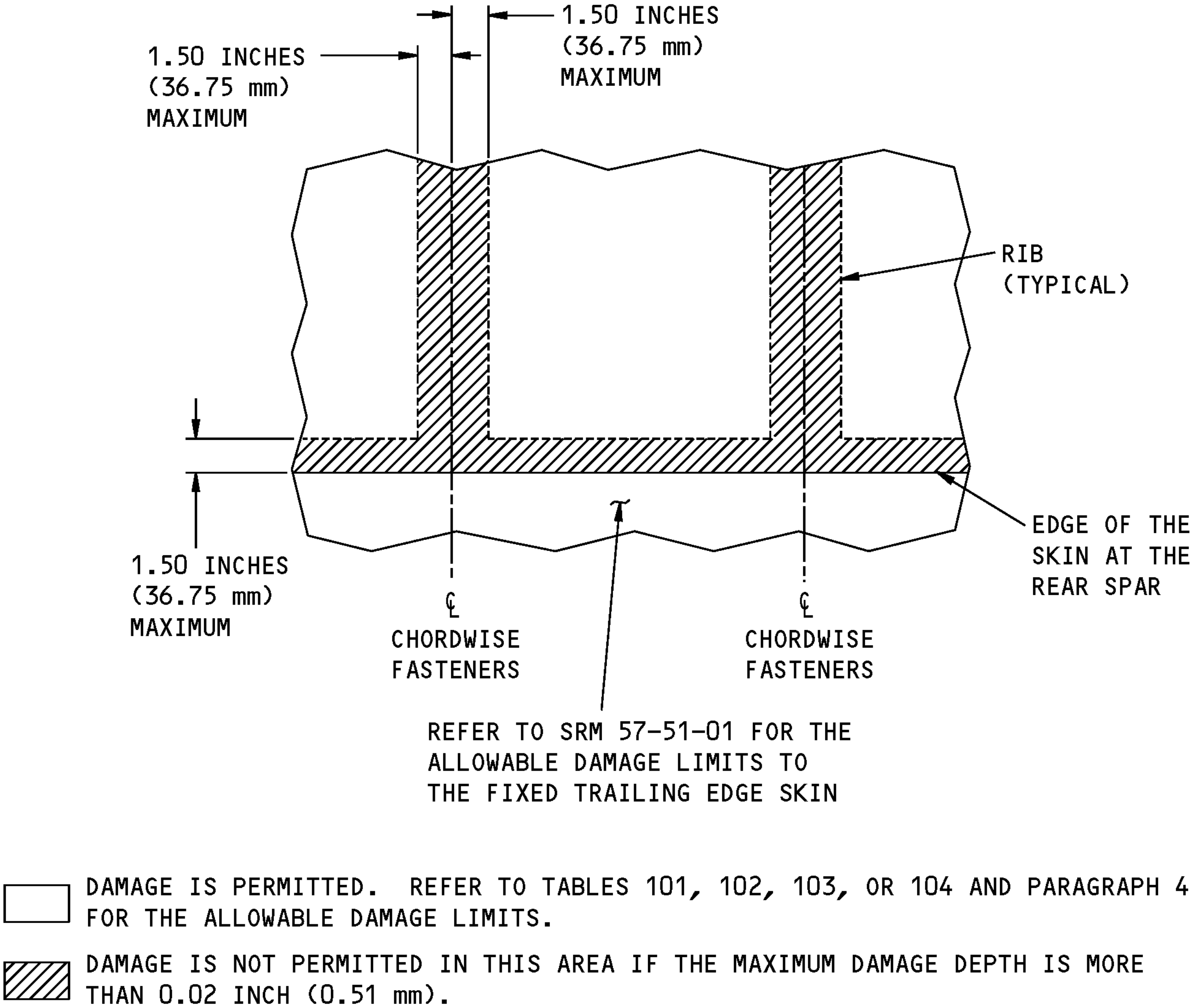
**Wing Inspar Skin Zones**
**Figure 102 (Sheet 1 of 4)**




LEFT SIDE LOWER SKIN IS SHOWN,
RIGHT SIDE IS OPPOSITE

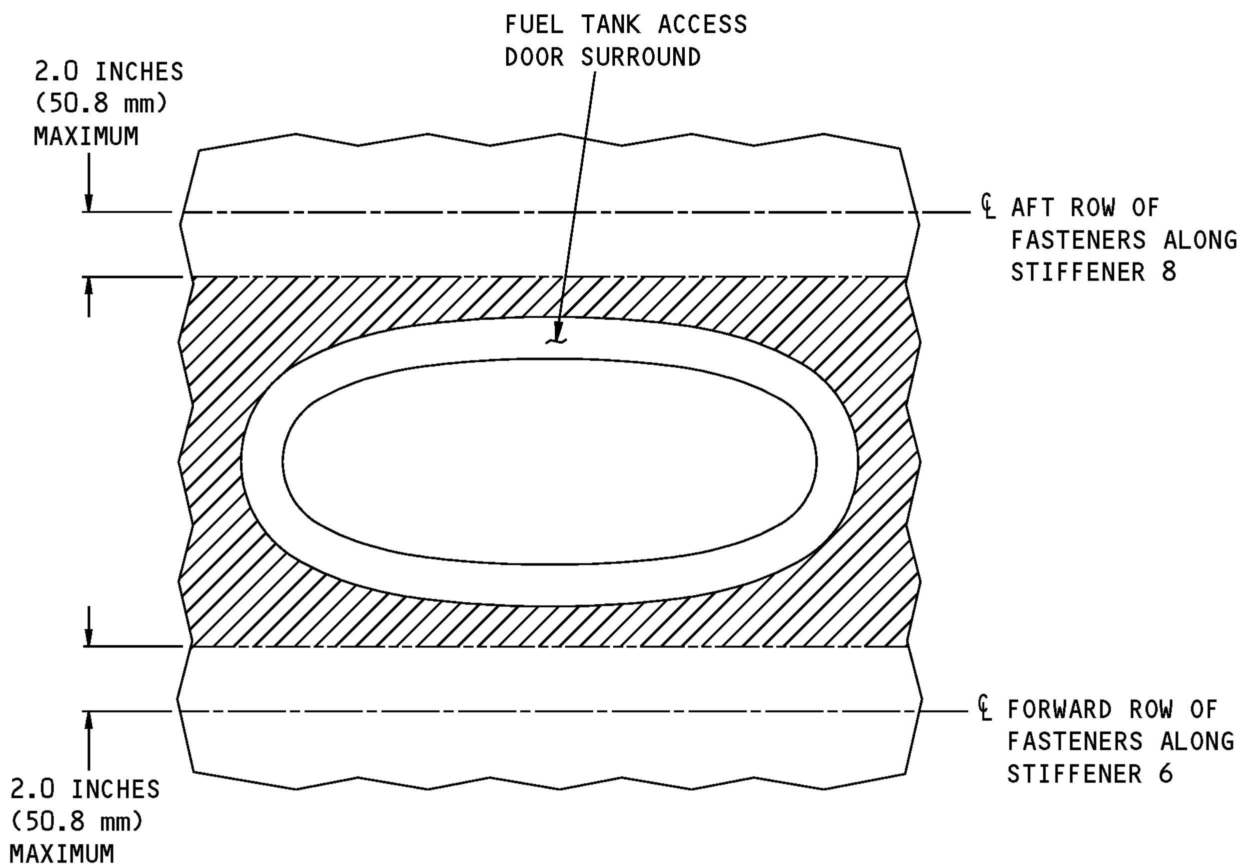
NOTE: THE LOWER INSPAR WING SKIN IS MADE OF MACHINED ALUMINUM.

**Wing Inspar Skin Zones**
**Figure 102 (Sheet 2 of 4)**




SKIN ZONES AT A REAR SPAR AND RIB AREA
(TYPICAL)
A

**Wing Inspar Skin Zones**
**Figure 102 (Sheet 3 of 4)**

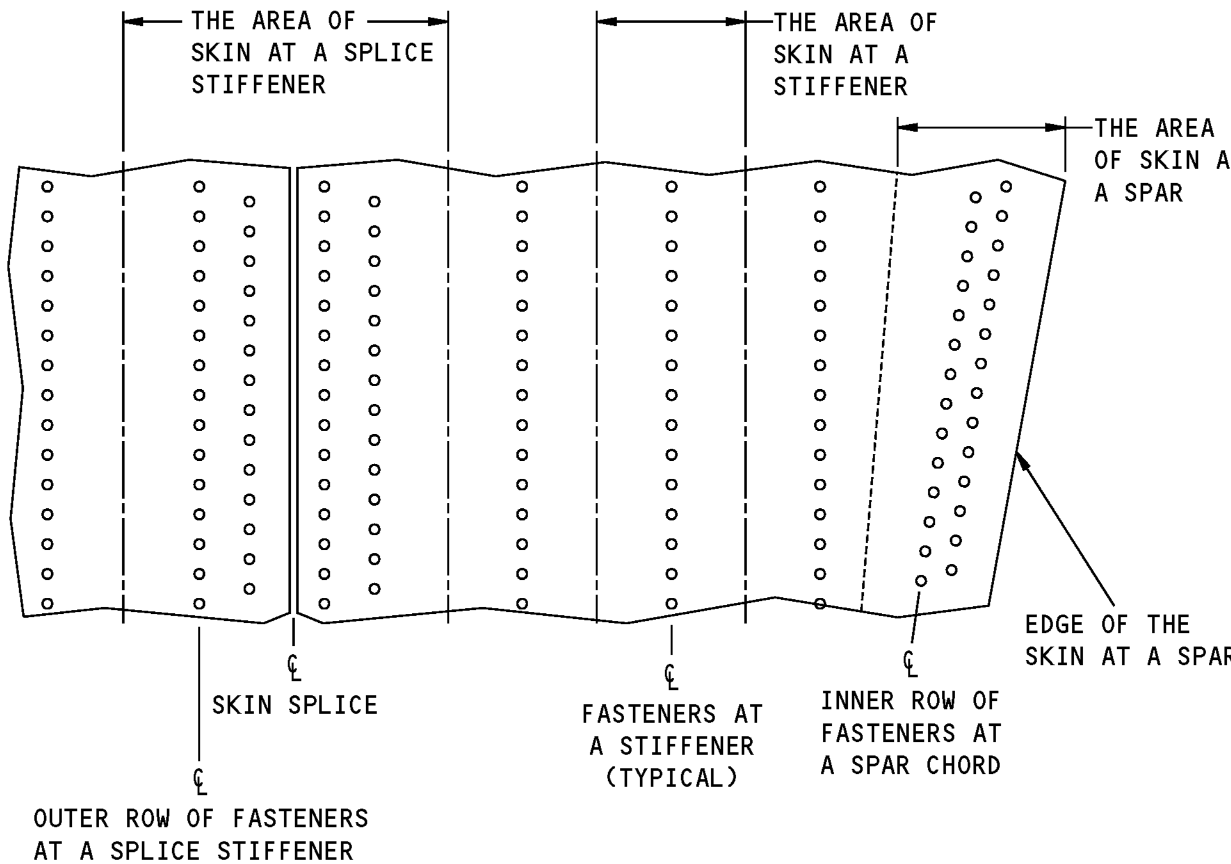



DAMAGE IS PERMITTED. REFER TO TABLE 103 OR TABLE 104 AND PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

DAMAGE IS NOT PERMITTED

LOWER SKIN ZONES BETWEEN STIFFENER 6 AND STIFFENER 8
(FUEL TANK ACCESS DOOR AREA IS SHOWN)
(TYPICAL)
B

**Wing Inspar Skin Zones**
**Figure 102 (Sheet 4 of 4)**




**Definitions of Skin Areas Used to Find the Allowable Damage Limits at a Stiffener or a Spar**
**Figure 103**

## 3. References

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| 51-40-09 | COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT |
| AMM 28-11-00 | Integral Fuel Tanks |
| AMM 28-11-11 | Wing Fuel Tank Access Panel Removal/Installation |
| AMM 28-23-00 | Fuel Tank Refueling |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-04-00 | Ultrasonic |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 4 | Surface Inspection of Aluminum Parts |

## 4. Allowable Damage Limits

A. Upper Skin (Except at Fastener Locations)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and G .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Lower Skin (Except at Fastener Locations)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , and H .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and G .

(b) Remove the damage at the fuel tank access or vent door surround as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , H , and I .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Upper and Lower Skin at a Fastener Location

(1) Refer to Table 106 to make sure you have a replacement fastener. Do not remove the damage if you do not have a replacement fastener.

(2) Remove the damage at an aluminum or titanium fastener.

**NOTE**: You must remove steel fasteners before you remove the damage. Do the procedures given in Paragraph 4.C.(3) for damage at a steel fastener.

(a) Sand or grind the aluminum fastener and the damaged skin around the fastener as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A .

**CAUTION:** DO NOT GRIND INTO A TITANIUM FASTENER. SMALL PARTICLES OF TITANIUM ARE FLAMMABLE. SAND INTO THE FASTENER MANUALLY OR WITH TOOLS. REFER TO SRM 51-10-02.

(b) Sand the titanium fastener and the damaged skin around the fastener as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(c) Do an inspection of the damage removal area.

1) Do a 10X visual inspection to make sure the damage is removed from under the surface.

2) Do an ultrasonic inspection to make sure the damage is removed from under the skin surface. Refer to 737 NDT Part 1, 51-04-00.

(3) If you have one of these three conditions, you must remove the fastener and the damage around the fastener area as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail B .

**NOTE**: Do not remove more than 75 percent of the fasteners at one time from skin attached to a structure.

(a) The fastener is made of steel.

(b) The damage is not fully removed from the aluminum or titanium fastener.

(c) The material removed from the aluminum or titanium fastener is more than is given in Table 105 .

(4) If, and after you remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail B , do an inspection of the area.

(a) Do an ultrasonic inspection to make sure the damage is removed from under the surface. Refer to 737 NDT Part 1, 51-04-00.

(b) Do a High Frequency Eddy Current (HFEC) inspection of the fastener hole to make sure there are no cracks. Refer to 737 NDT Part 6, 51-00-00, Figure 4 .

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

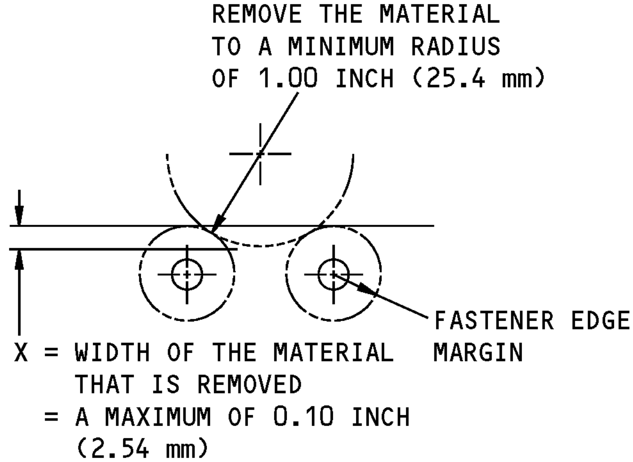
(5) Shot peen the damage area.
   (a) Refer to 51-20-06 for the shot peen intensity and shot number.
   (b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(6) If you removed the fastener, do as follows.
   (a) Drill and countersink the fastener hole.
   (b) Cold Work the fastener if it is in the lower skin.
      1) Use the High Interference Cold Work method. Refer to 51-40-09.
   (c) Replace the fastener you removed with the applicable fastener given in Table 106 or Table 107 .

**Table 107**:

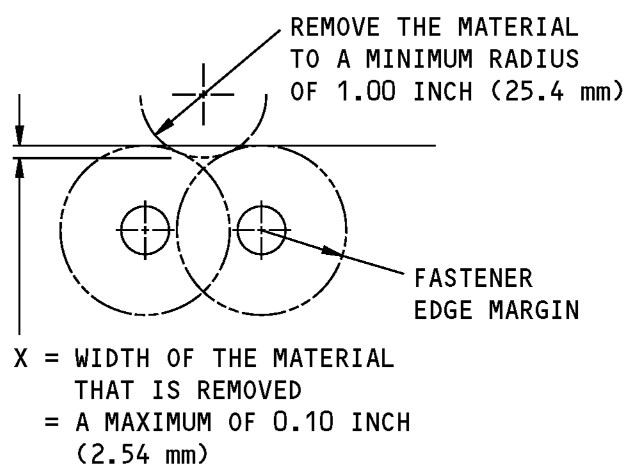
| INITIAL FASTENER | FASTENER | MINIMUM THICKNESS OF THE SKIN THAT THAT REMAINS AFTER CLEANUP | REPLACEMENT FASTENER |
|---|---|---|---|
| ALUMINUM RIVET WITH 82°/30° COUNTERSINK | 3/16 | 0.08 | BACB30NW6K()Y BOLT WITH A BACC30AC6 COLLAR |
| | 1/4 | 0.09 | BACB30NW8K()Y BOLT WITH A BACC30R8 COLLAR |
| | 5/16 | 0.10 | BACB30NW10K()Y BOLT WITH A BACC30R10 COLLAR |
| | 3/8 | 0.12 | BACB30NW12K()Y BOLT WITH A BACC30R12 COLLAR |

**NOTE**: Use the data in this table only if the minimum thickness of the skin that remains after cleanup is less than is given in Table 106 for the initial fastener given.




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

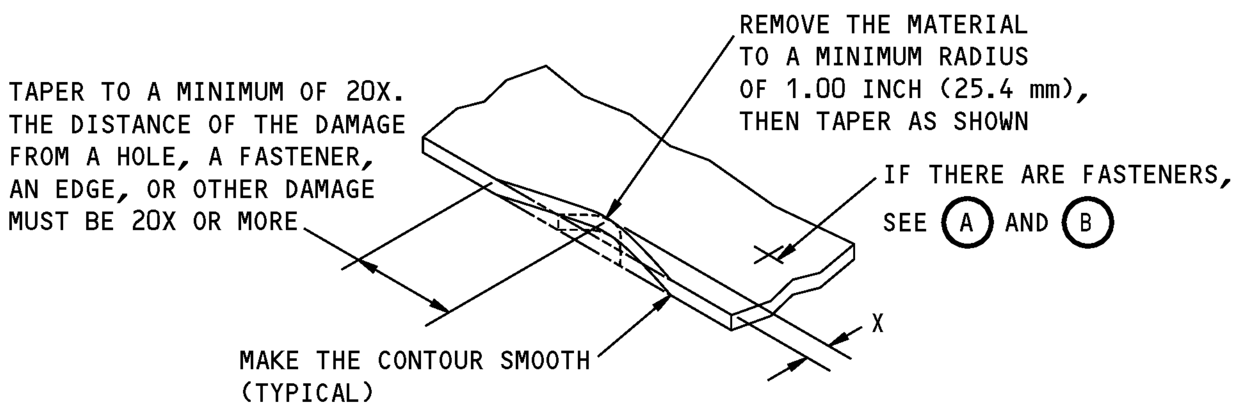
A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 8)**

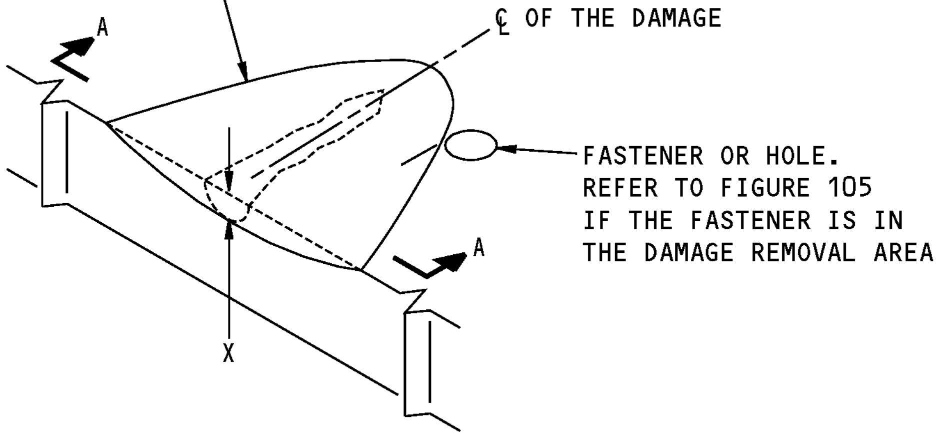



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH (2.54 mm)

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN

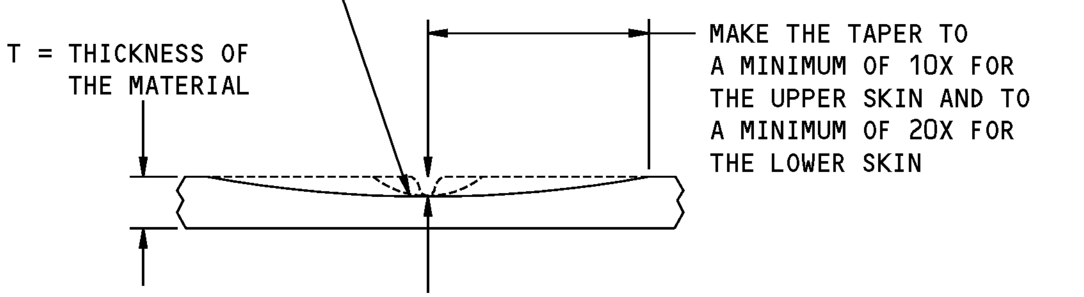
C

**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 8)**




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

D



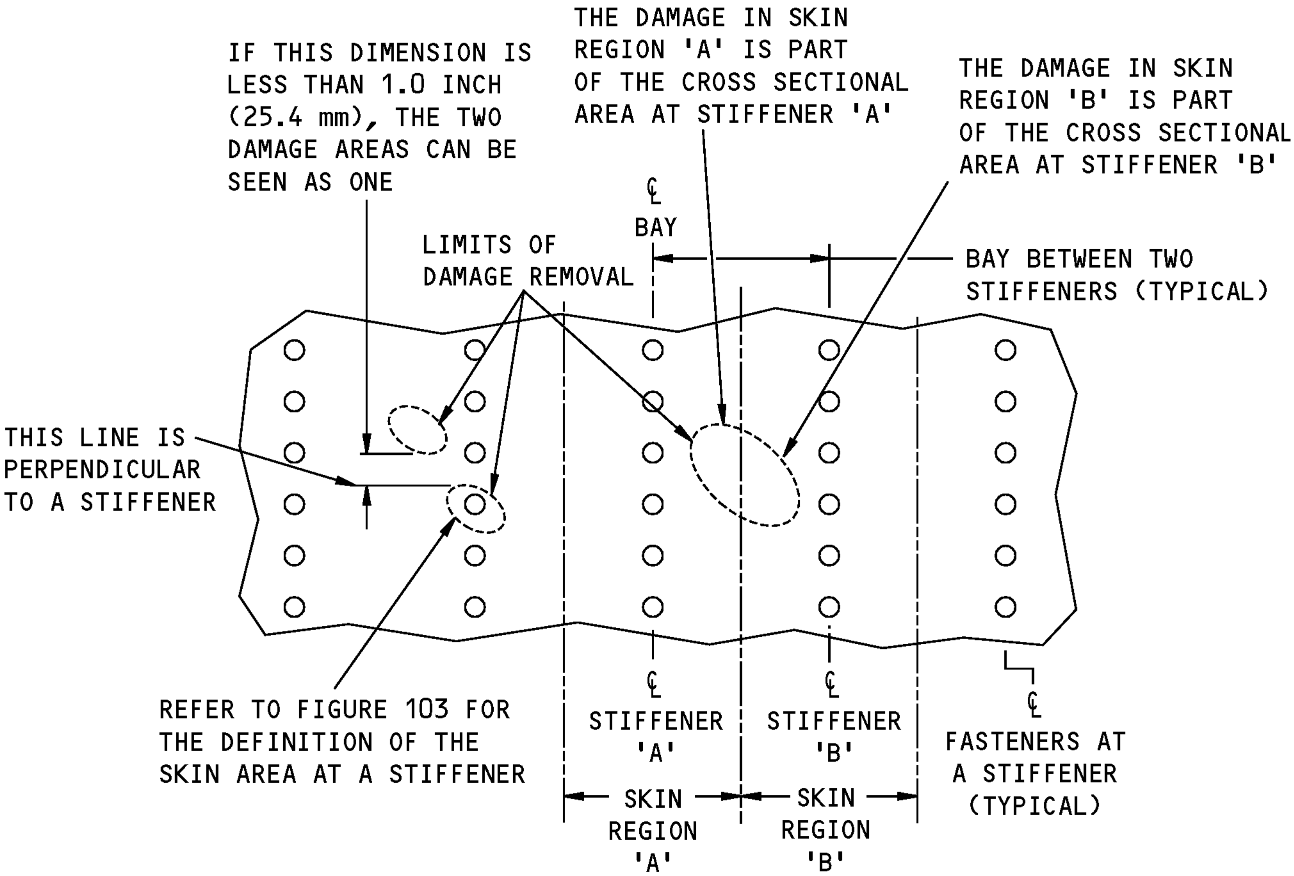

X = DEPTH OF THE MATERIAL
THAT IS REMOVED
= A MAXIMUM OF 0.05 INCH (1.3 mm) FOR
CHEM-MILLED SKIN AT THE DOOR SURROUND
OF THE FUEL TANK ACCESS AND VENT DOORS

(REFER TO TABLE 101 OR TABLE 102 FOR THE
UPPER SKIN)

(REFER TO TABLE 103 OR TABLE 104 FOR THE
LOWER SKIN)

A-A

**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 8)**

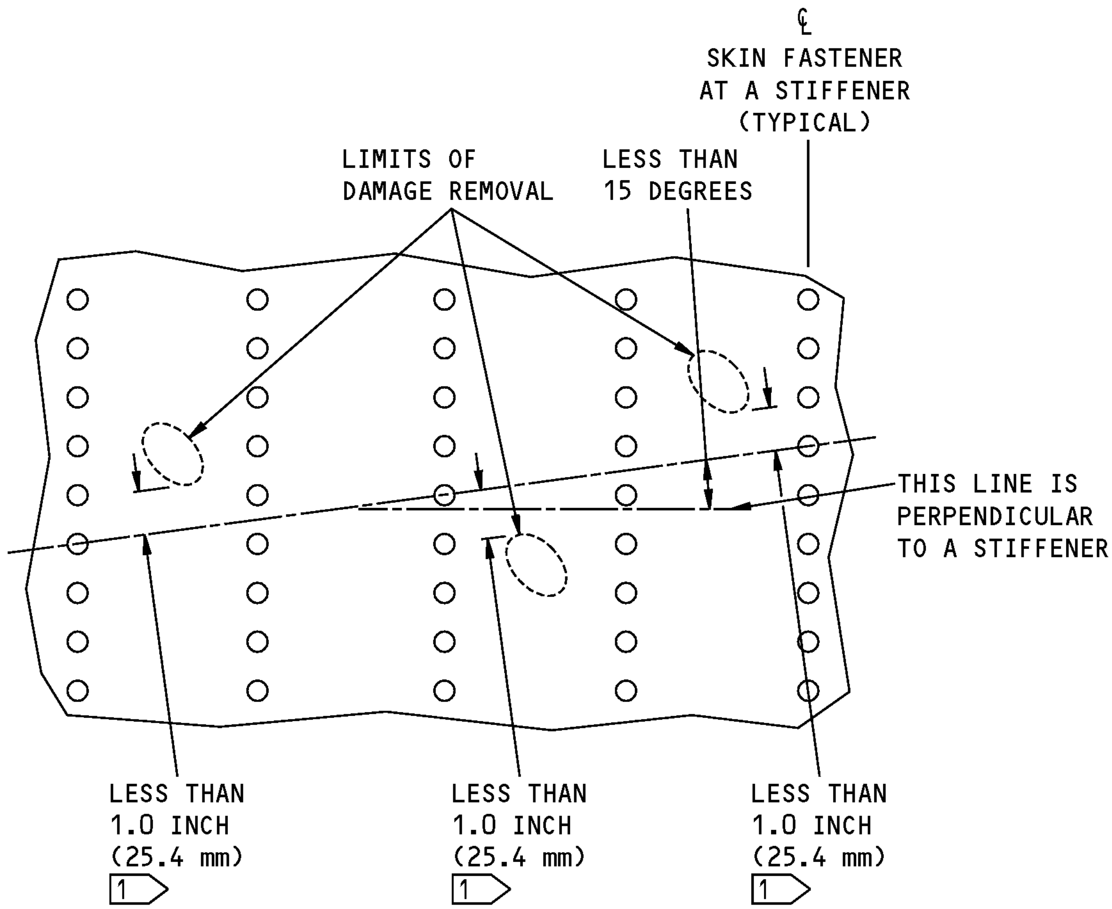



NOTE: REFER TO FIGURE 104, DETAIL G AND PARAGRAPH 5, TO CALCULATE THE CROSS SECTIONAL AREA THAT IS REMOVED.

QUANTITY OF DAMAGE IN A CROSS SECTIONAL AREA AT A STIFFENER OR A SPAR

E

**Allowable Damage Limits**
**Figure 104 (Sheet 4 of 8)**



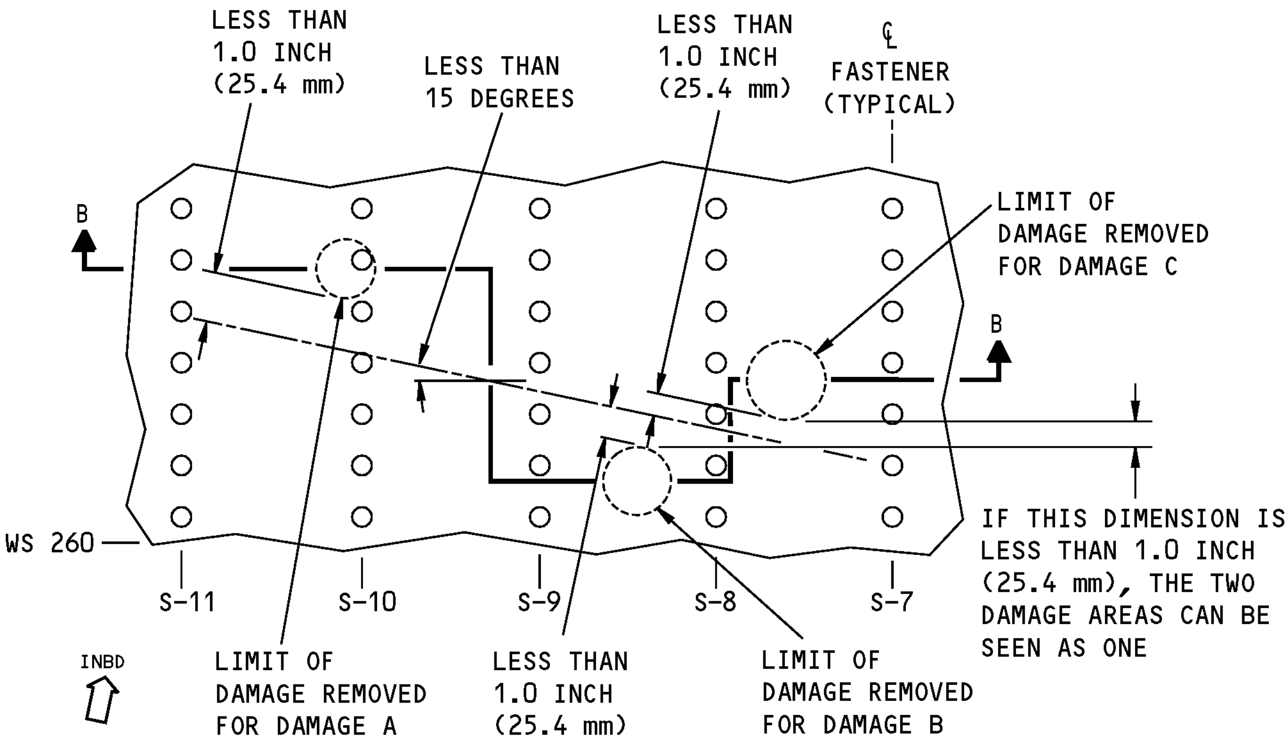

NOTE: REFER TO FIGURE 104, DETAIL G AND PARAGRAPH 5, TO CALCULATE THE CROSS SECTIONAL AREA THAT IS REMOVED.

1 INCLUDE THIS DAMAGE AS THE TOTAL DAMAGE FROM THE FRONT SPAR TO THE REAR SPAR.

QUANTITY OF DAMAGE BETWEEN THE FRONT SPAR AND THE REAR SPAR

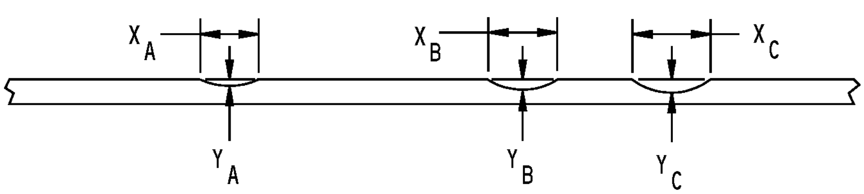
F

**Allowable Damage Limits**
**Figure 104 (Sheet 5 of 8)**



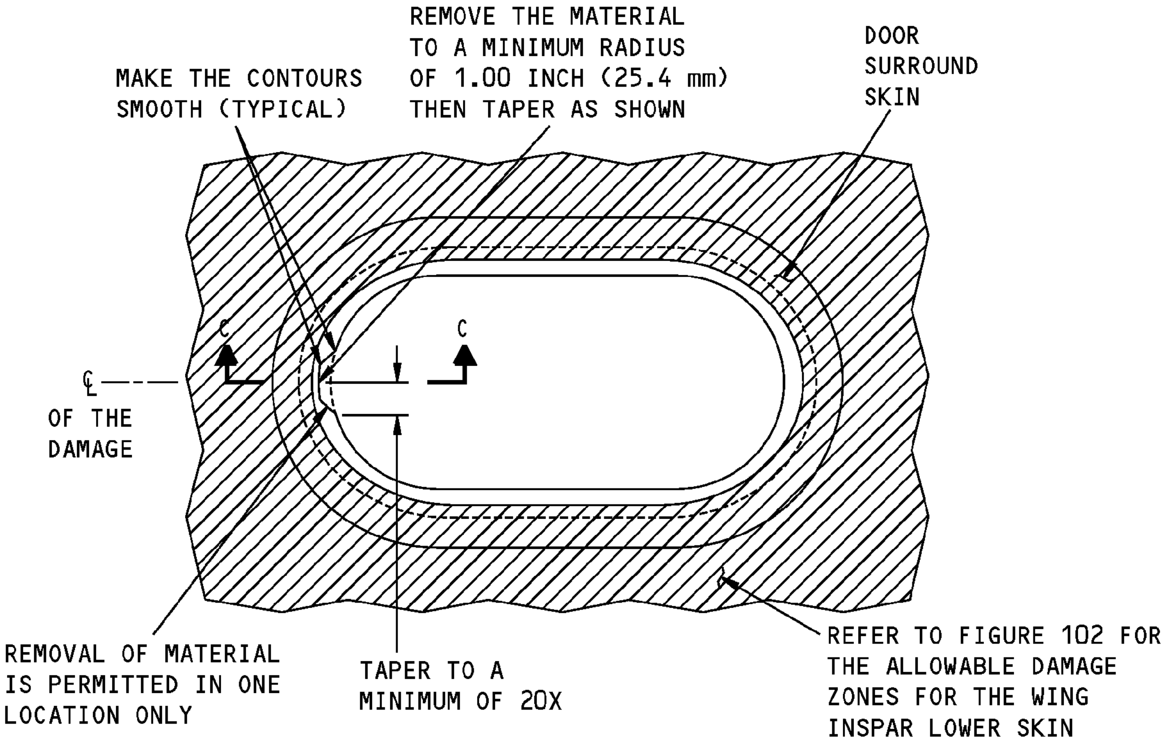

REMOVAL OF DAMAGED MATERIAL AT STIFFENERS AND BETWEEN SPARS
(REFER TO PARAGRAPH 5 FOR AN EXAMPLE)

G



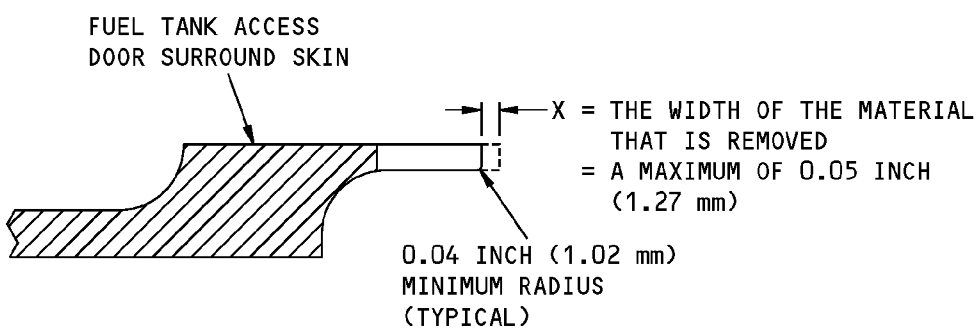

B-B

**Allowable Damage Limits**
**Figure 104 (Sheet 6 of 8)**




REMOVAL OF DAMAGED MATERIAL AT THE EDGE OF
A FUEL TANK ACCESS OR VENT DOOR SURROUND SKIN

H

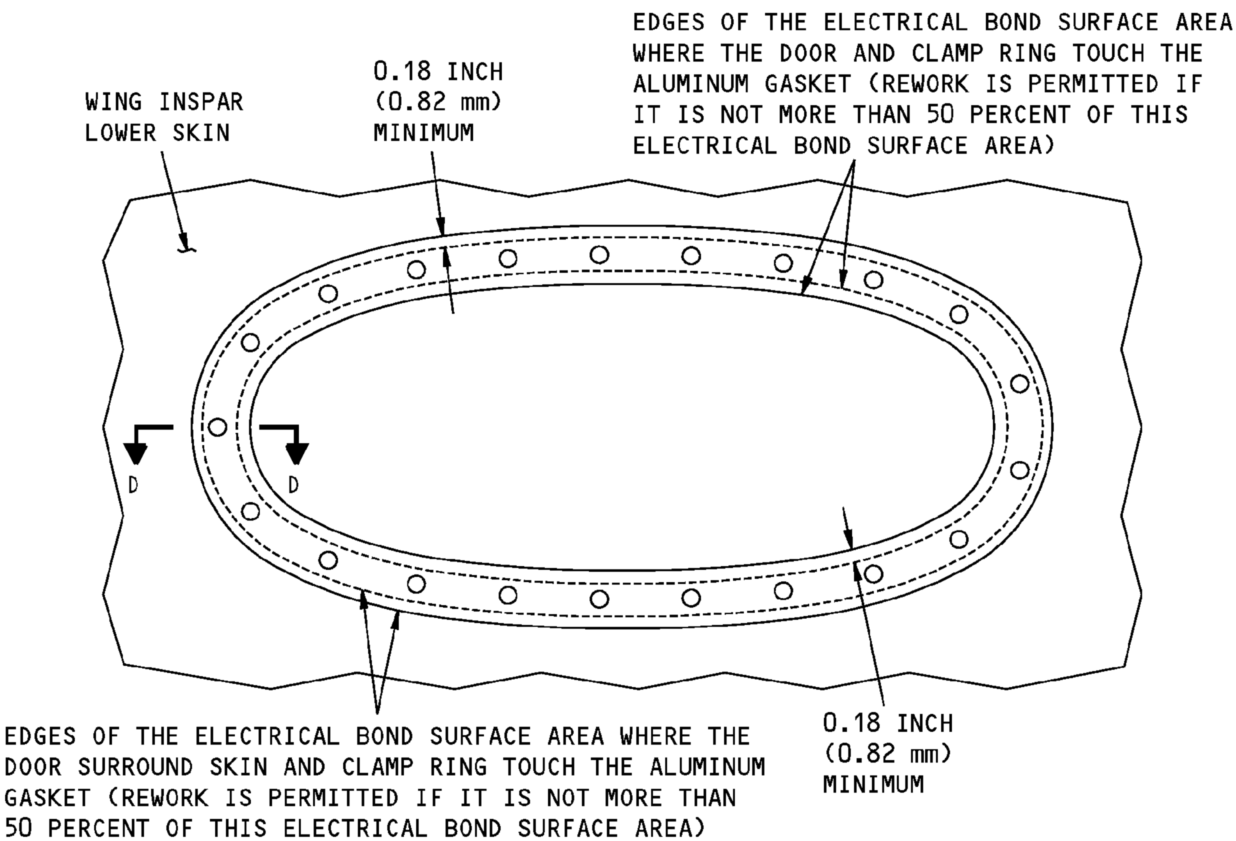



DAMAGE IS PERMITTED AS SHOWN.

DAMAGE IS NOT PERMITTED.

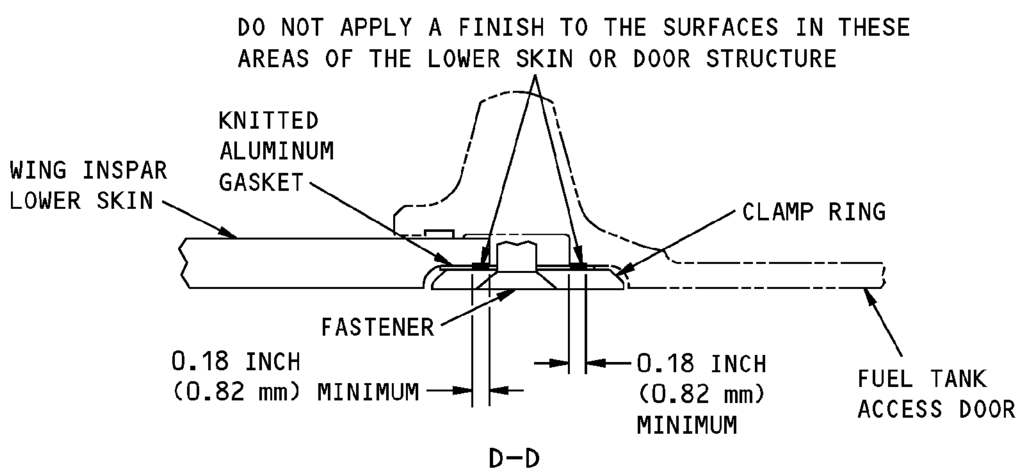
C-C

**Allowable Damage Limits**
**Figure 104 (Sheet 7 of 8)**



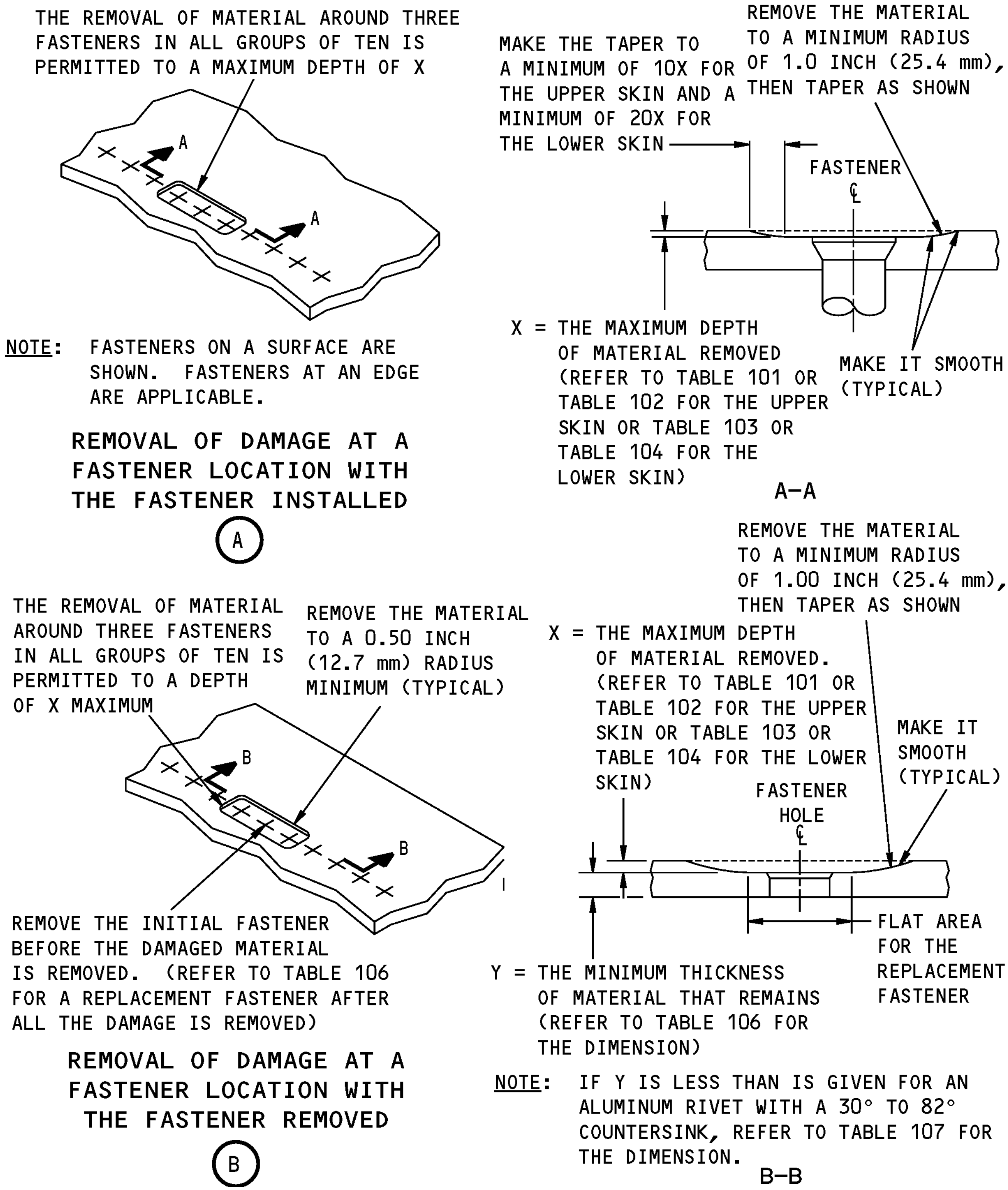

ELECTRICAL BOND LIMITS FOR FUEL TANK ACCESS
OR VENT DOOR SURROUND SKIN, CLAMP RING, AND DOOR STRUCTURE

I




D-D

**Allowable Damage Limits**
**Figure 104 (Sheet 8 of 8)**




**Allowable Damage Limits**
**Figure 105**

5. **Example to Calculate the Cross-Sectional Area Removed**

A. Calculate the damage depths and areas shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail G to find if the damage is permitted.

**NOTE**: You can make a conservative estimate of the area loss as described below if you can not accurately calculate the area loss. In the example below, the cross sectional area loss is estimated to be 1/2 the width X depth.

(1) Find the allowable damage depth (Y) for each damage location shown.

(a) The damage depth at Damage A is 0.01 inch (0.25 mm).

1) This is less than 0.02 inch (0.51 mm) and is permitted.

(b) The damage depth at Damage B is 0.01 inch (0.25 mm).

1) This is less than 0.02 inch (0.51 mm) and is permitted.

(c) The damage depth at Damage C is 0.015 inch (0.38 mm).

1) This is less than 0.02 inch (0.51 mm) and is permitted.

(2) Calculate the cross sectional area that is decreased between the stringers, then find if the decreased area is permitted.

(a) The area that has decreased at Damage A is:

1) 2.20 inches/2 X 0.01 inch = 0.01 inch$^2$ (7.10 mm$^2$)

a) This is less than 0.02 inch$^2$ (12.90 mm$^2$) and is permitted.

(b) The area that has decreased at Damage B is:

1) 2.40 inches/2 X 0.01 inch = 0.012 inch$^2$ (7.74 mm$^2$)

a) This is less than 0.02 inch$^2$ (12.90 mm$^2$) and is permitted.

(c) The area that has decreased at Damage C is:

1) 2.50 inches/2 X 0.015 inch = 0.019 inch$^2$ (12.26 mm$^2$)

a) This is less than 0.02 inch$^2$ (12.90 mm$^2$) and is permitted.

(3) Calculate the total cross sectional area that has decreased between the front and rear spars. After you do this, find if that decreased area is permitted.

(a) Add the areas that have decreased in Damage A, B, and C as the total area decreased between the front and rear spars.

1) [0.01 + 0.012 + 0.019] inch$^2$ = 0.041 inch$^2$ (26.45 mm$^2$).

a) This is less than 0.20 inch$^2$ (129.03 mm$^2$) and is permitted.

## ALLOWABLE DAMAGE 2 - WING INSPAR SKIN OUTBOARD OF WBL 616.75

1. **Applicability**

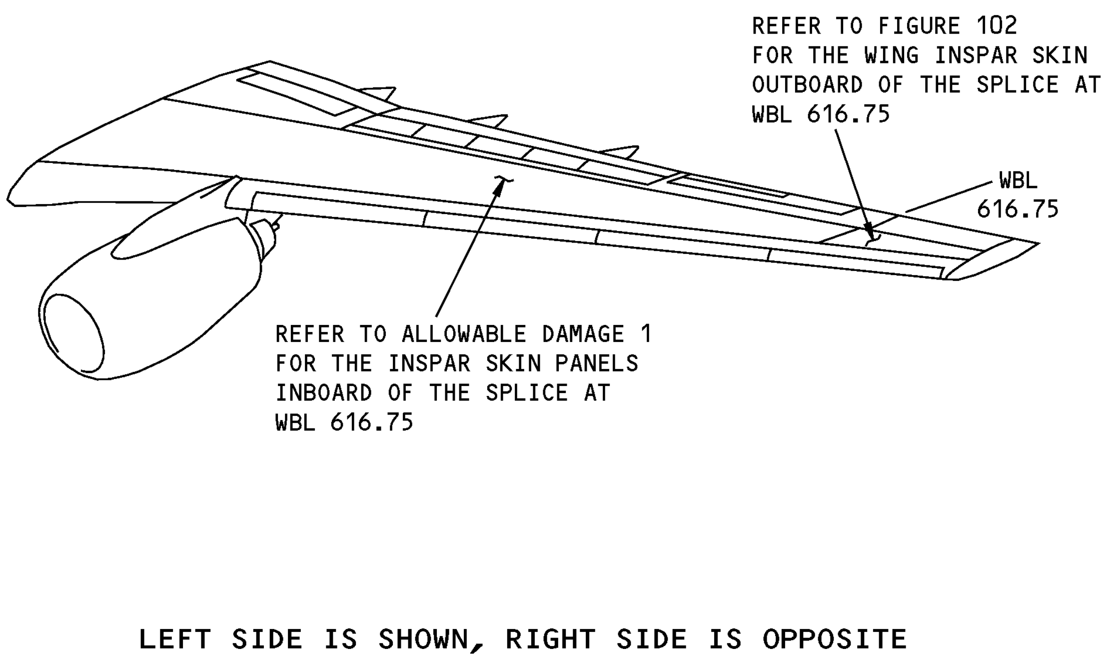
   A. This subject gives the allowable damage limits for the wing inspar skin outboard of WBL 616.75 shown in Wing Inspar Skin Location, Figure 101/ALLOWABLE DAMAGE 2.

   B. Allowable Damage 2 is applicable to airplanes that have not had winglets installed.

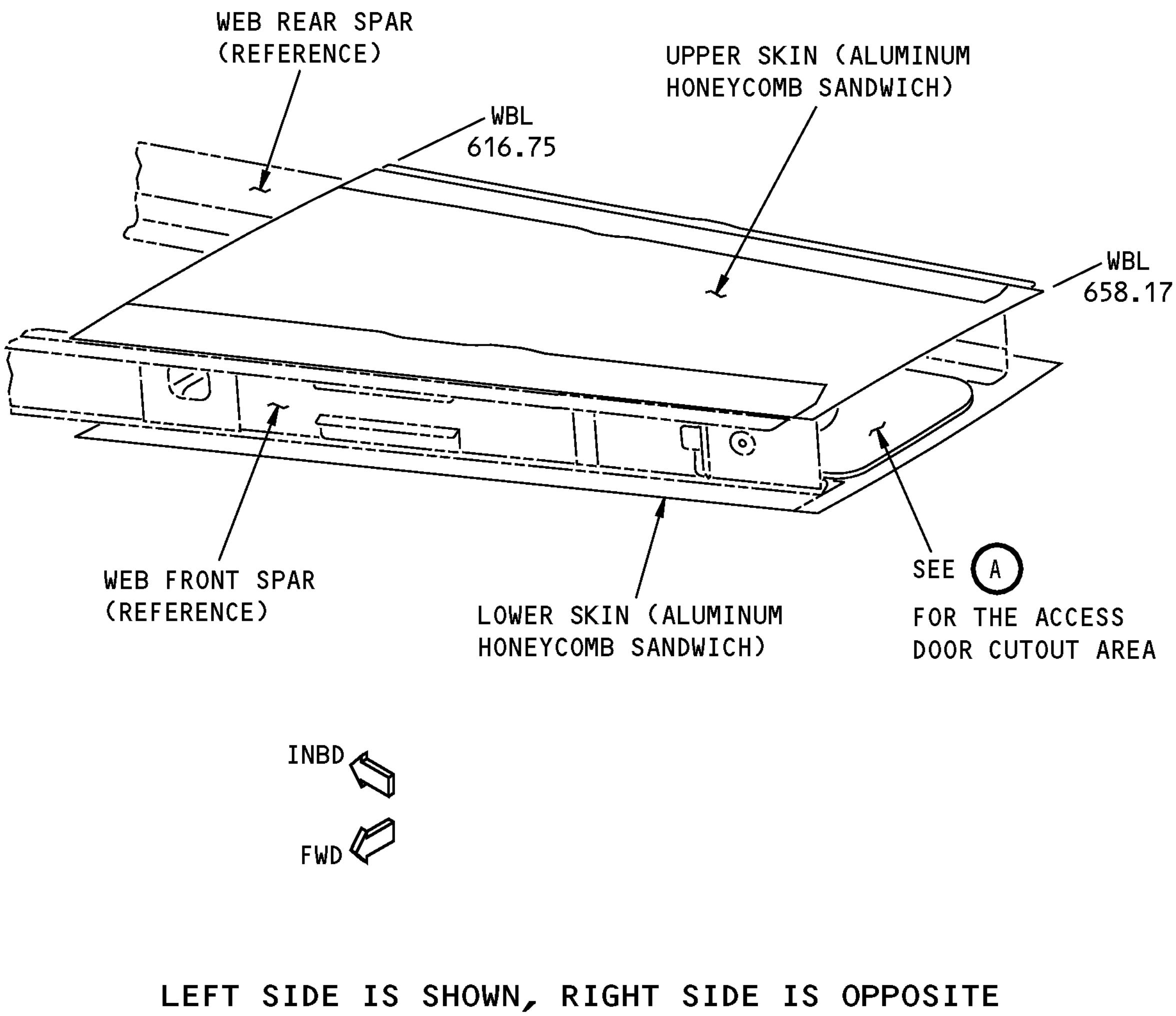
   C. Allowable Damage 2 is not applicable to:

      (1) Airplanes that have had winglets installed as part of the production of the airplane. Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets.

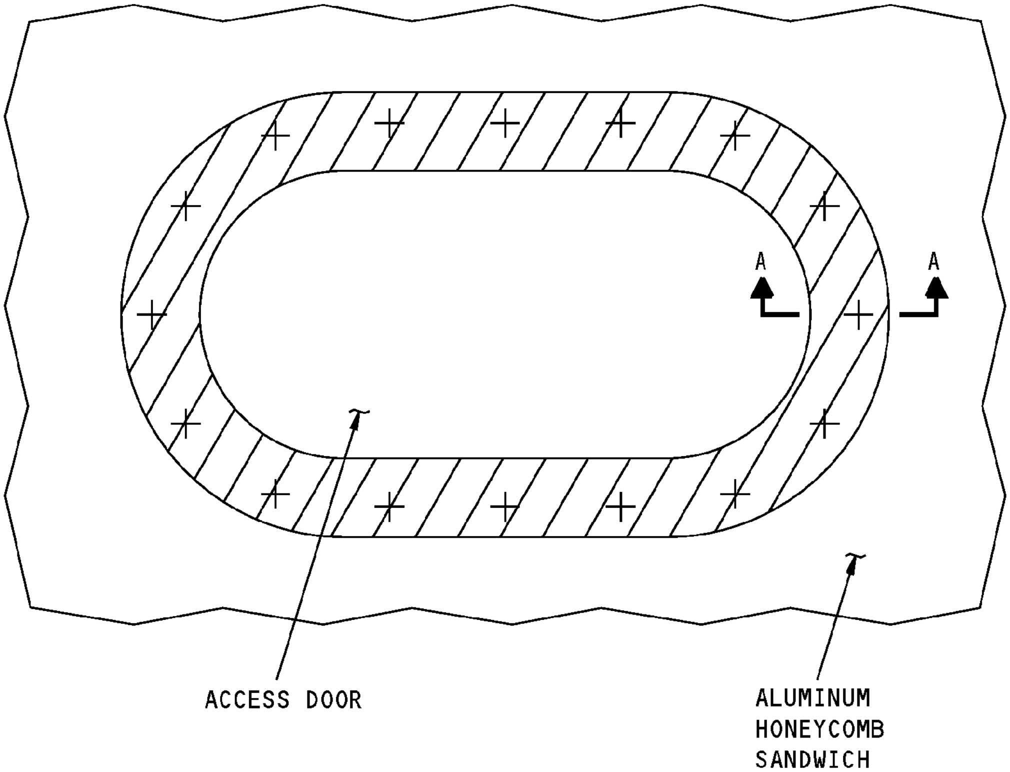
      (2) Airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.




**Wing Inspar Skin Location**
**Figure 101**

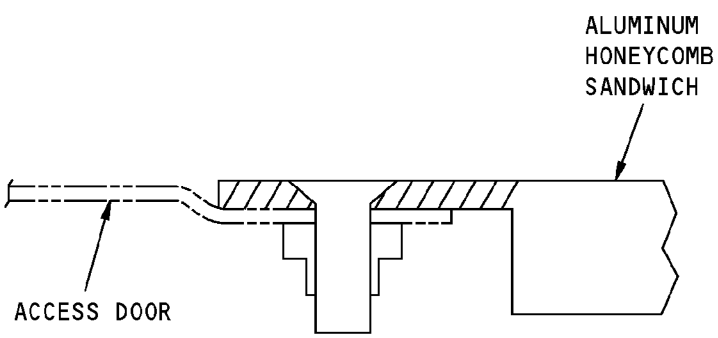



**Wing Inspar Skin Outboard of the Splice at WBL 616.75**
**Figure 102 (Sheet 1 of 2)**




THE ACCESS DOOR SURROUND IN THE LOWER HONEYCOMB SKIN IS SHOWN

A




A-A

ACCESS DOOR SURROUND (ALUMINUM SKIN)

**Wing Inspar Skin Outboard of the Splice at WBL 616.75**
**Figure 102 (Sheet 2 of 2)**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

(1) Apply a chemical conversion coating to the bare surfaces of the skin. Refer to 51-20-01.

(2) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked skins. Refer to SOPM 20-41-02.

(3) Apply a layer of BMS 10-79, Type III primer to the surfaces of the reworked skins. Refer to SOPM 20-44-04.

3. **References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-10 | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. **Allowable Damage Limits**

A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and G .

1) Only one damage removal area is permitted for each 15 square inches (9679 square mm) of skin surface area.

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail E if:

1) It is not more than 15 square inches (9679 square mm) for each square foot (0.09 square m) of skin surface area

2) The distance away from a hole, other damage, or part edge is 4D (D= the largest dimension of the damage).

(b) If the damage is more than the above conditions, then you are permitted to fill or rework the dent as given in 51-70-01.

1) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval, or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted in the outer facesheet of the honeycomb area up to 0.25 inch (6.35 mm) in diameter if:

1) It is a minimum of 1.0 inch (25.4 mm) away from a hole, other damage, or part edge

2) There is not more than one damage area for each 15 square inches (9679 square mm) of skin surface area

3) You make a temporary seal

a) Apply aluminum foil tape (speed tape).

b) Keep a record of the location.

c) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

d) Repair the damage at or before 5000 flight hours from the time the seal was made.

(b) The damage is not permitted in the edgebands.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail F .

B. Lower Skin - Access Door Surround (Aluminum Skin)

(1) Cracks:

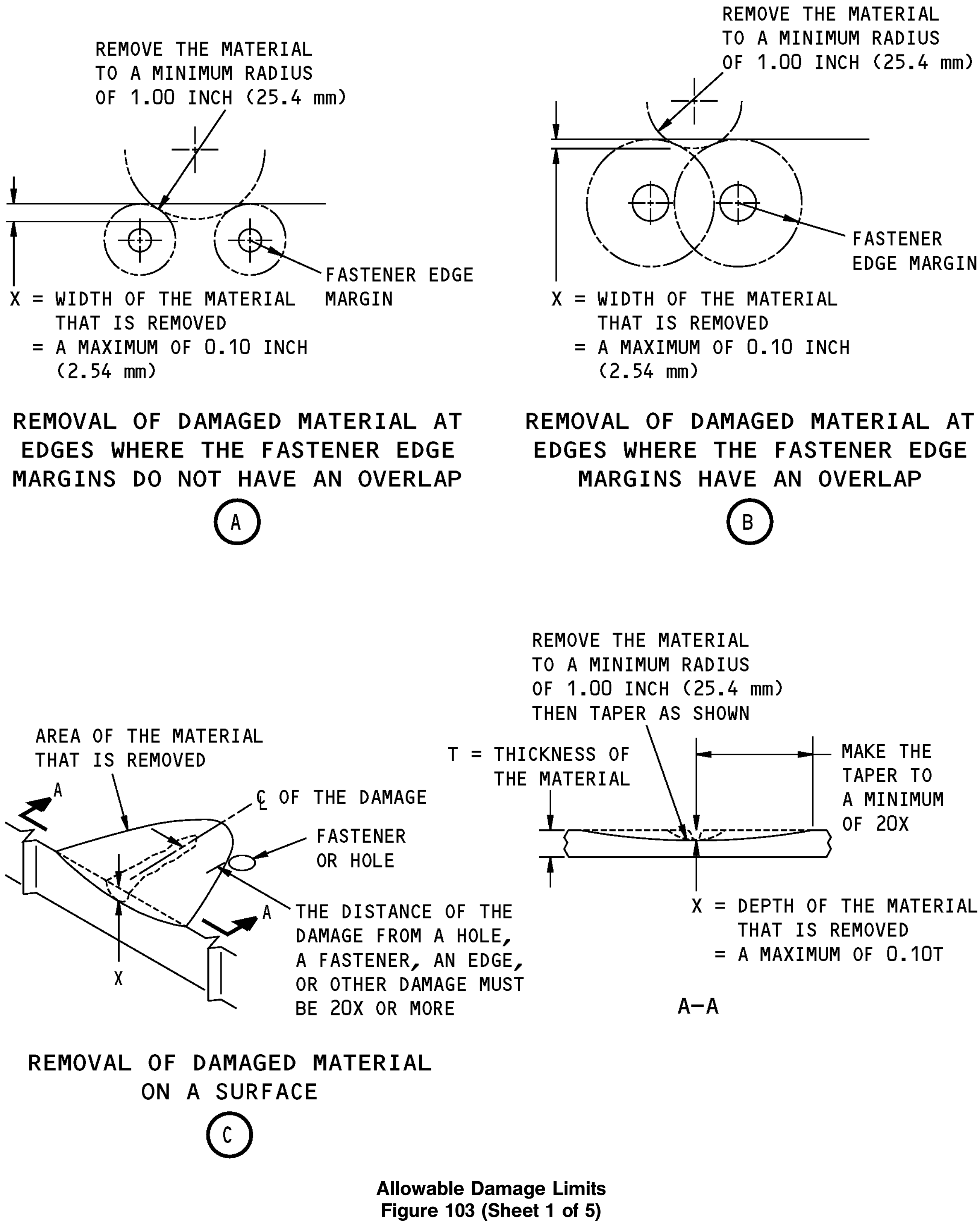
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail H .

(2) Nicks, Gouges, Scratches, and Corrosion:

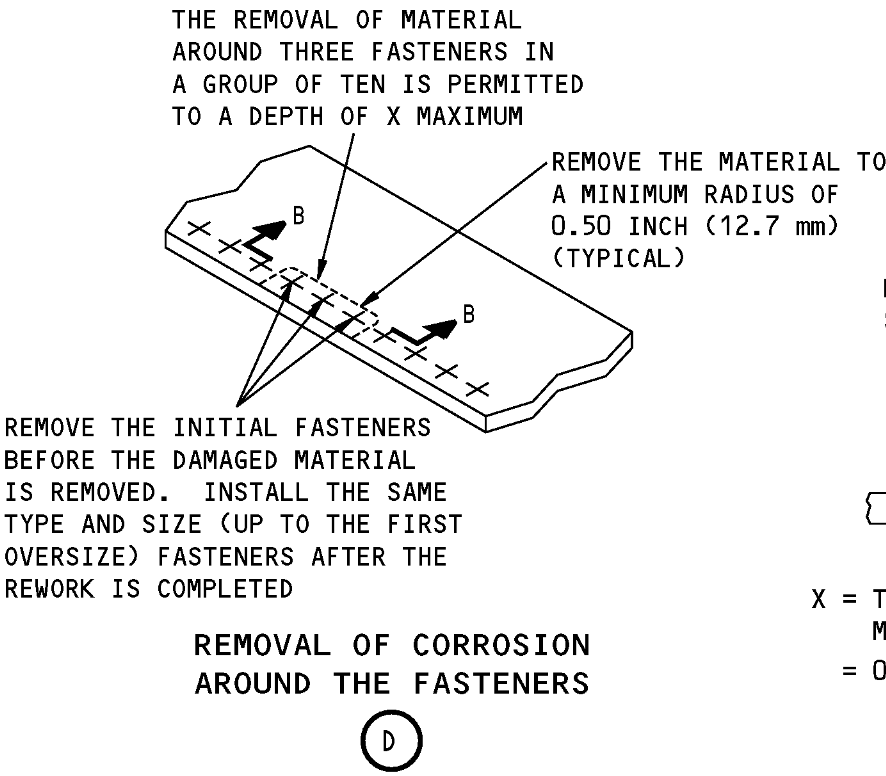
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details C and H .

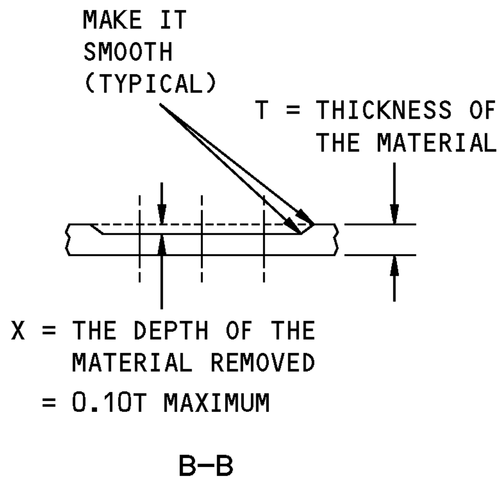
(3) Dents are not permitted.

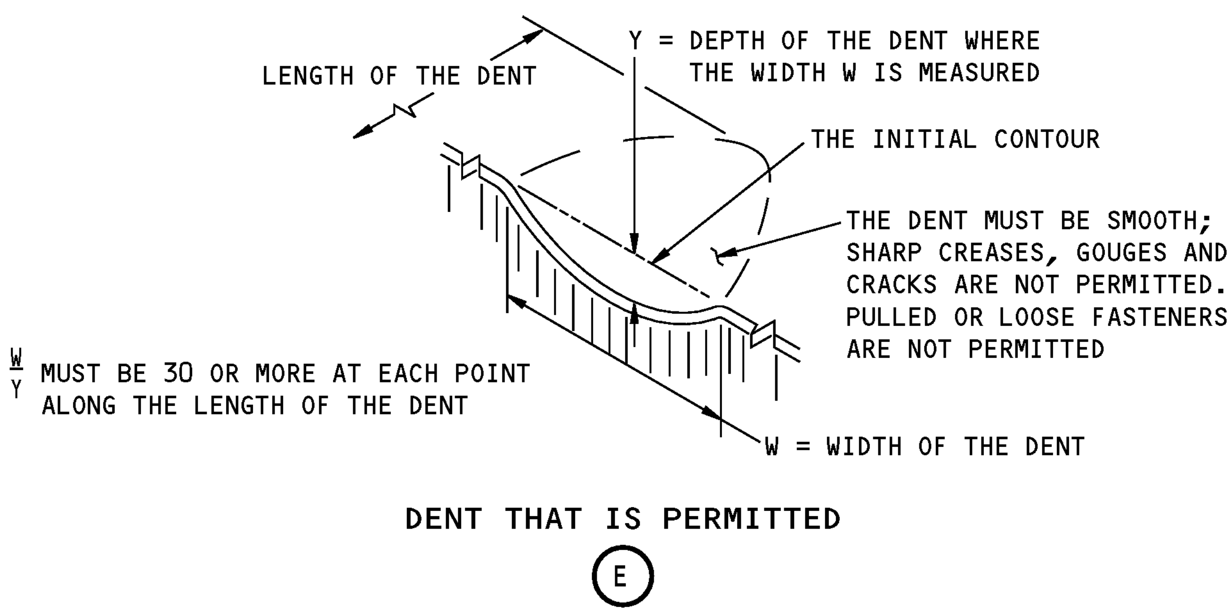
(4) Holes and Punctures are not permitted.



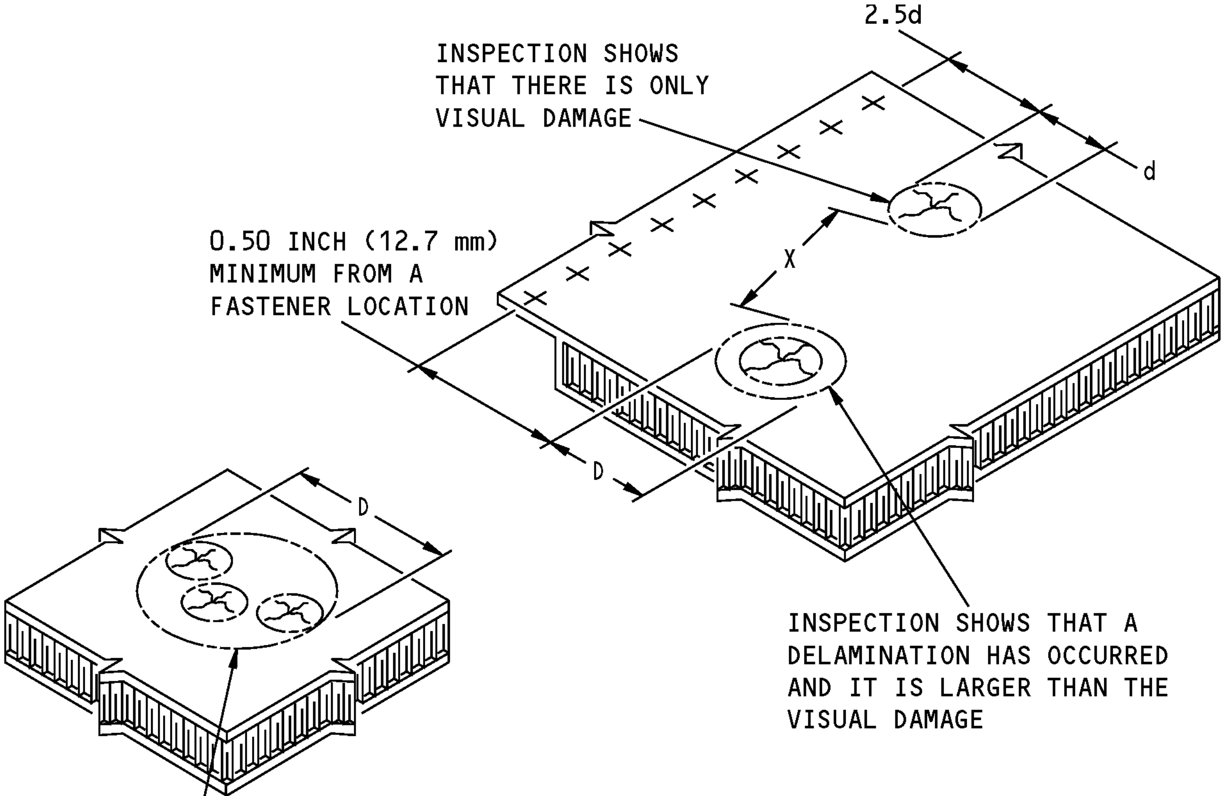

**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 5)**




**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 5)**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 2 INCHES (50.8 mm).

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

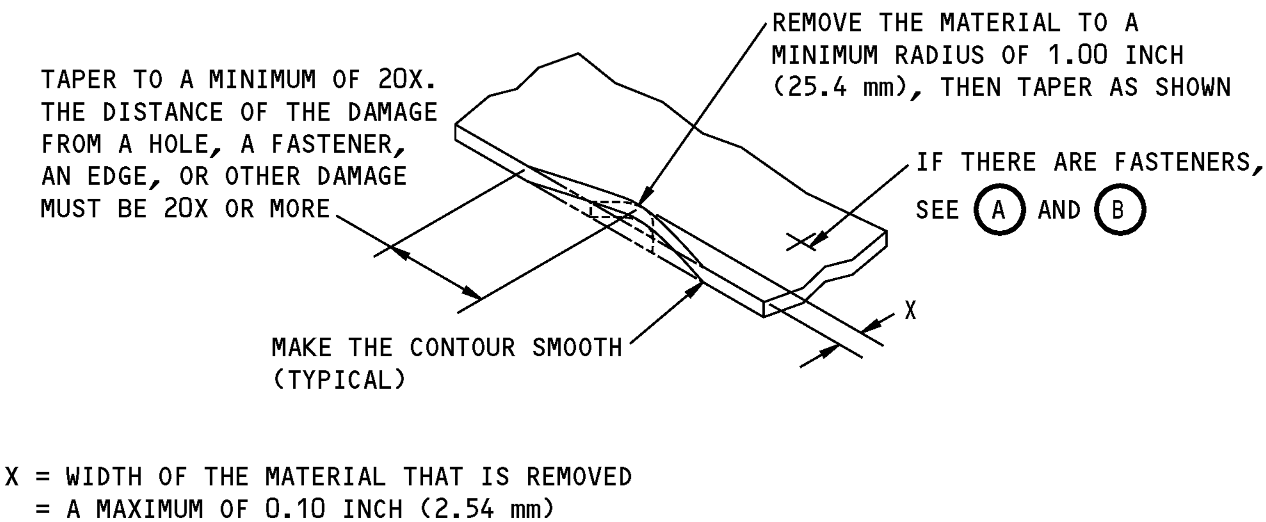
X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 3d.

DAMAGE THAT IS PERMITTED TO HONEYCOMB PANELS

F

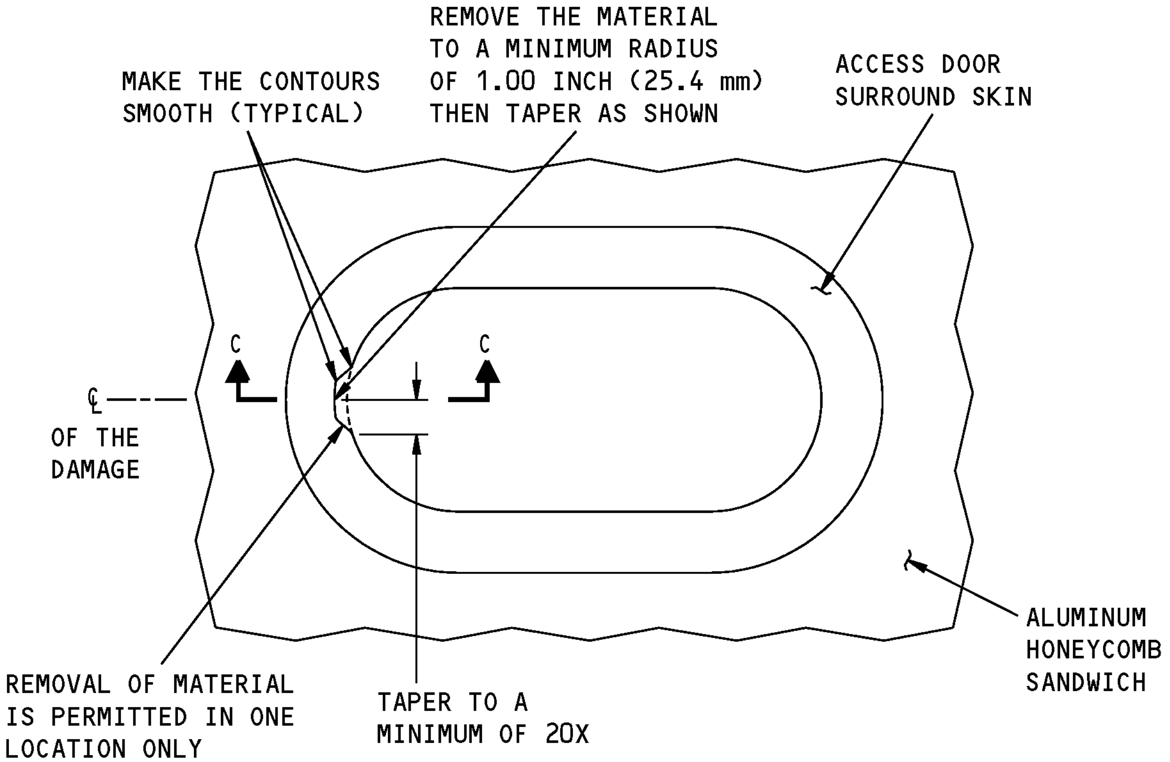
**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 5)**




REMOVAL OF DAMAGED MATERIAL
AT AN EDGE OF A METAL SKIN

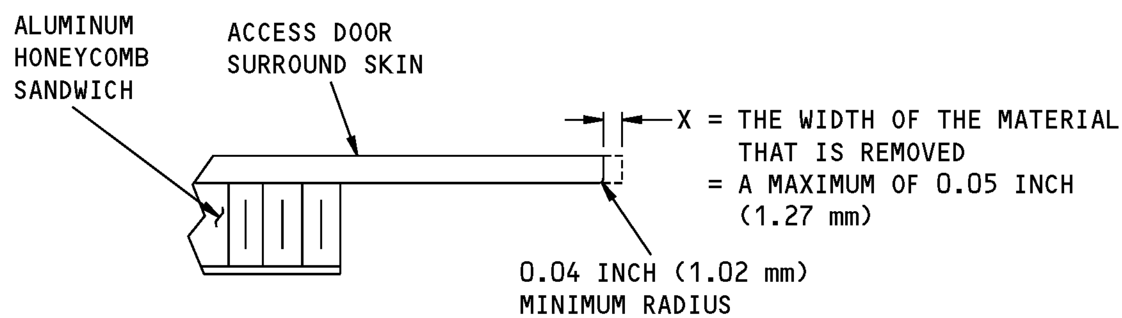
G

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 5)**




REMOVAL OF DAMAGED MATERIAL AT THE EDGE OF THE ACCESS DOOR SURROUND

H

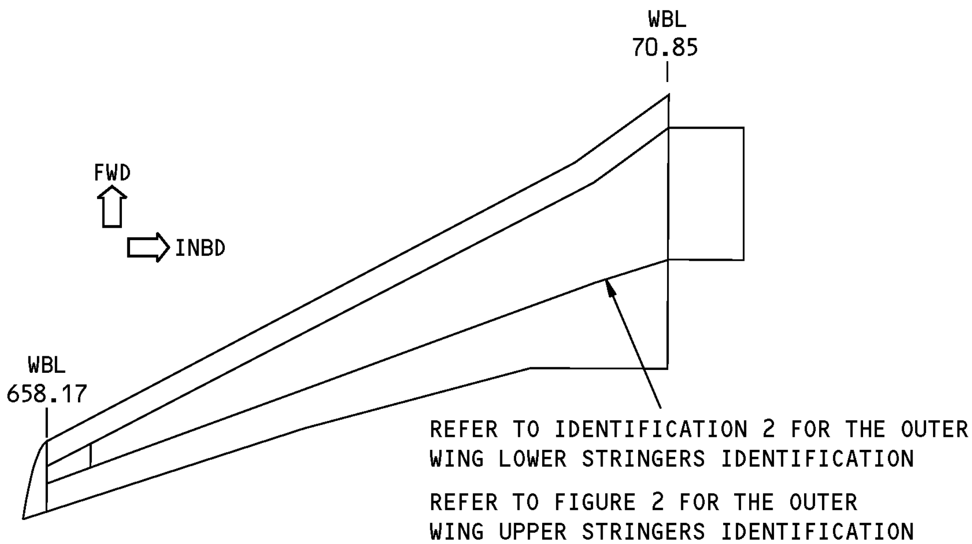



TYPICAL BLENDOUT AT ACCESS DOOR SURROUND
C-C

**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 5)**

## IDENTIFICATION 1 - OUTER WING UPPER STRINGERS

### 1. Locator




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Outer Wing Upper Stringers Location**
**Figure 1**

### 2. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 110A1503 | Wing Stringer Diagram, Upper 737-NG |
| 112A3001 | Panel Installation - Upper, Section 12, Wing |

### 3. Applicability

A. Identification 1 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 1 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

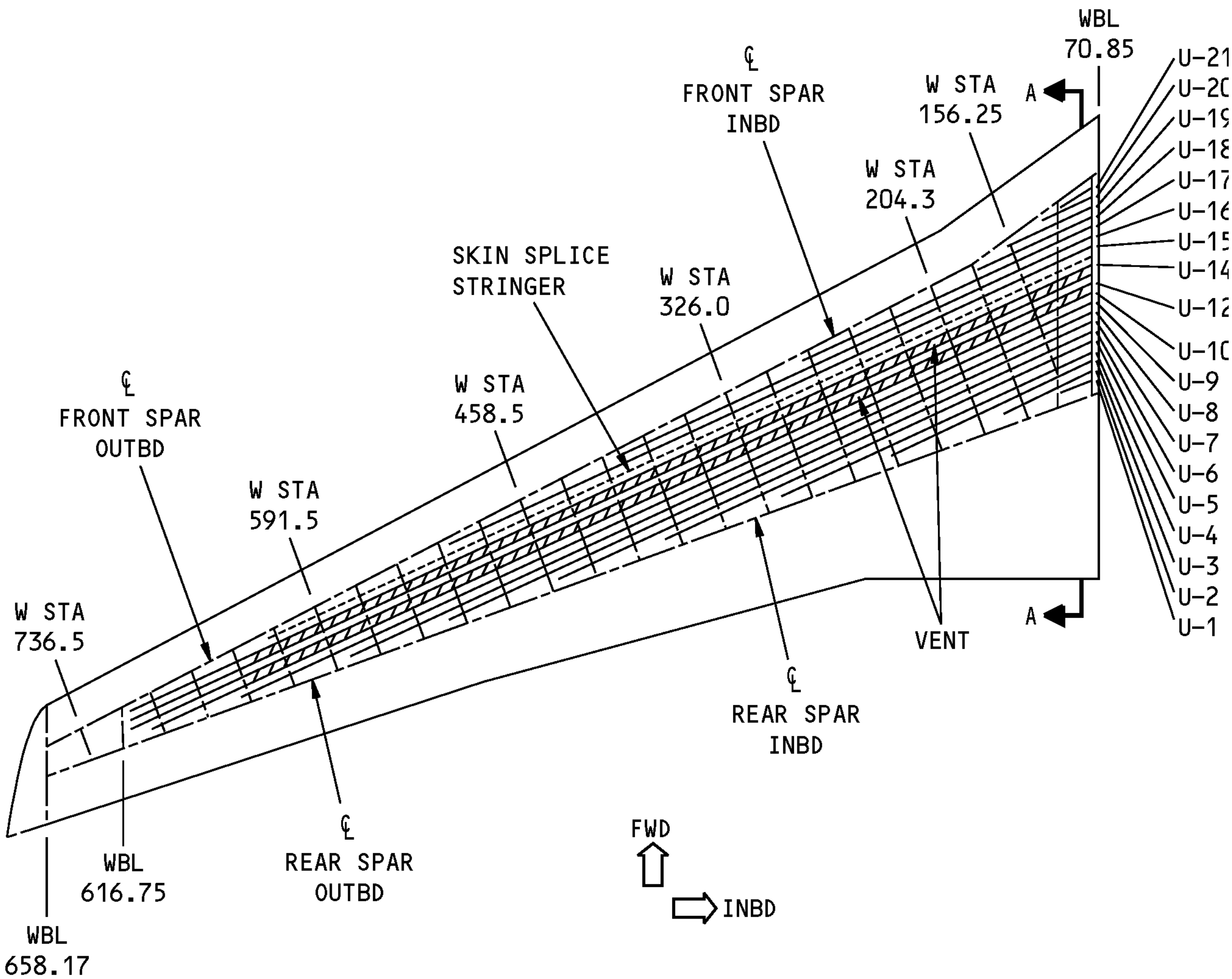
1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. List of Materials

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| STRINGER | DESCRIPTION | MATERIAL | DRAWING NUMBER | EFFECTIVITY |
| U-1 | Splice | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3201 | |
| U-2 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3202 | |
| U-3 | Zee | BAC1518-1169 7055-T77511 Extrusion as given in BMS 7-308. | 112A3203 | |
| U-4 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3204 | |
| U-5 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3205 | |
| U-6 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3206 | |
| U-7 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3207 | |
| U-8 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3208 | |
| U-9 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3209 | |
| U-10 and U-11 | Vent | BAC1510-1313 7055-T77511 Extrusion as given in BMS 7-308. | 112A3210 | Cum Line 1 thru 777 |
| | Vent | BAC1510-1378 7055-T77511 Extrusion as given in BMS 7-308. | 112A3210 | Cum Line 778 and on |
| U-12 and U-13 | Vent | BAC1510-1313 7055-T77511 Extrusion as given in BMS 7-308. | 112A3212 | Cum Line 1 thru 777 |
| | Vent | BAC1510-1378 7055-T77511 Extrusion as given in BMS 7-308. | 112A3212 | Cum Line 778 and on |
| U-12 and U-13 | Vent | BAC1510-1313 7055-T77511 Extrusion as given in BMS 7-308 (Optional: BAC1510-1378) | 112A3212 | |
| U-14 | Splice | BAC1518-1169 7055-T77511 Extrusion as given in BMS 7-308. | 112A3214 | |
| U-15 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3215 | |
| U-16 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3216 | |
| U-17 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3217 | |
| U-18 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3218 | |
| U-19 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3219 | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| STRINGER | DESCRIPTION | MATERIAL | DRAWING NUMBER | EFFECTIVITY |
| U-20 | Zee | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3220 | |
| U-21 | Splice | BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308. | 112A3221 | |

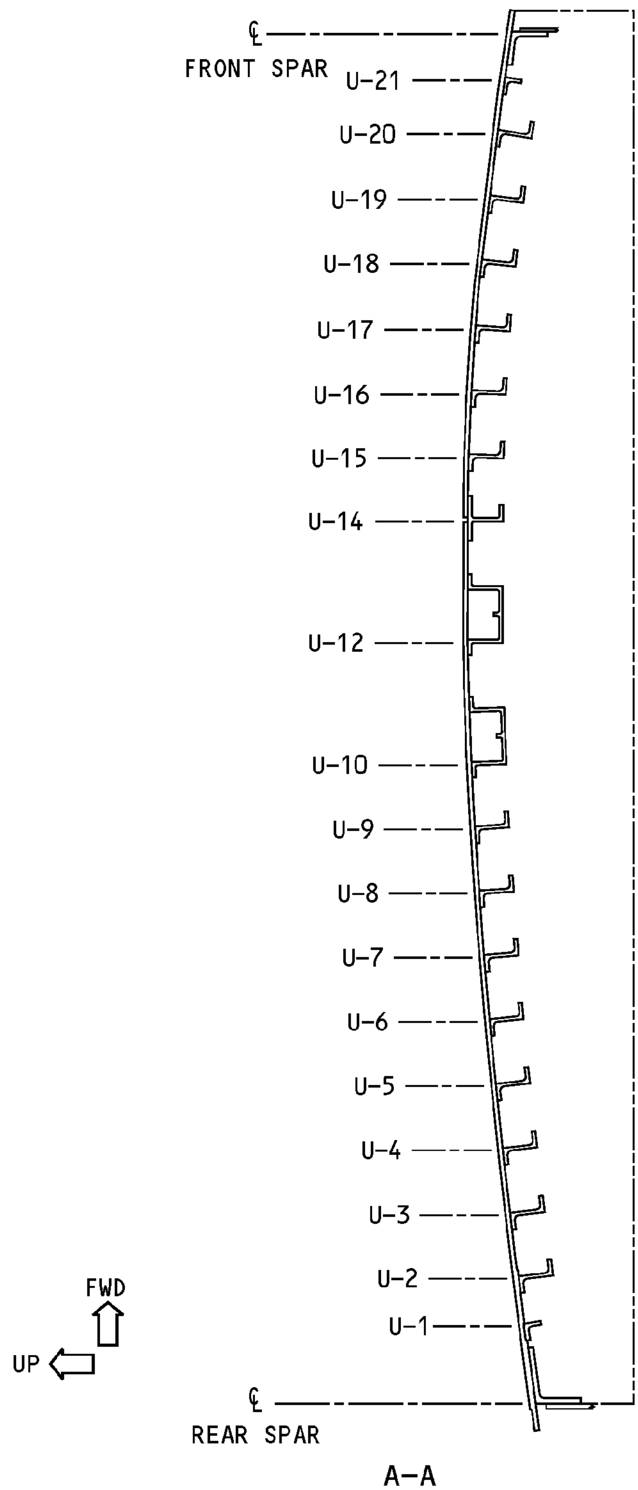



LEFT WING IS SHOWN, RIGHT WING IS OPPOSITE
PLAN VIEW OF OUTER WING UPPER SURFACE

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

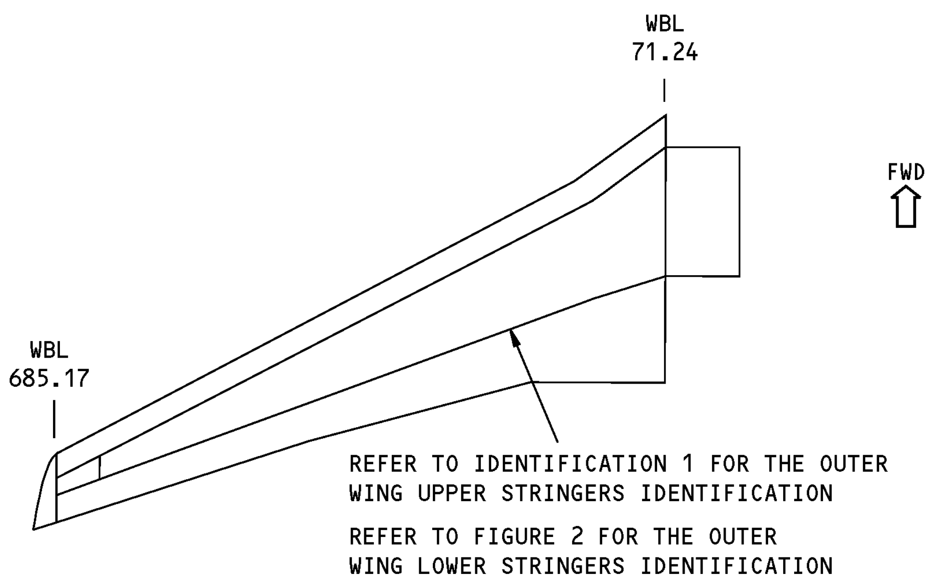
**Outer Wing Upper Stringers Identification**
**Figure 2 (Sheet 1 of 2)**




**Outer Wing Upper Stringers Identification**
**Figure 2 (Sheet 2 of 2)**

## IDENTIFICATION 2 - OUTER WING LOWER STRINGERS

### 1. Locator




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Outer Wing Lower Stringers Location**
**Figure 1**

### 2. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 110A1504 | Wing Stringer Diagram - Lower 737-NG |
| 112A4001 | Panel Installation - Lower, Section 12, Wing |

### 3. Applicability

A. Identification 2 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

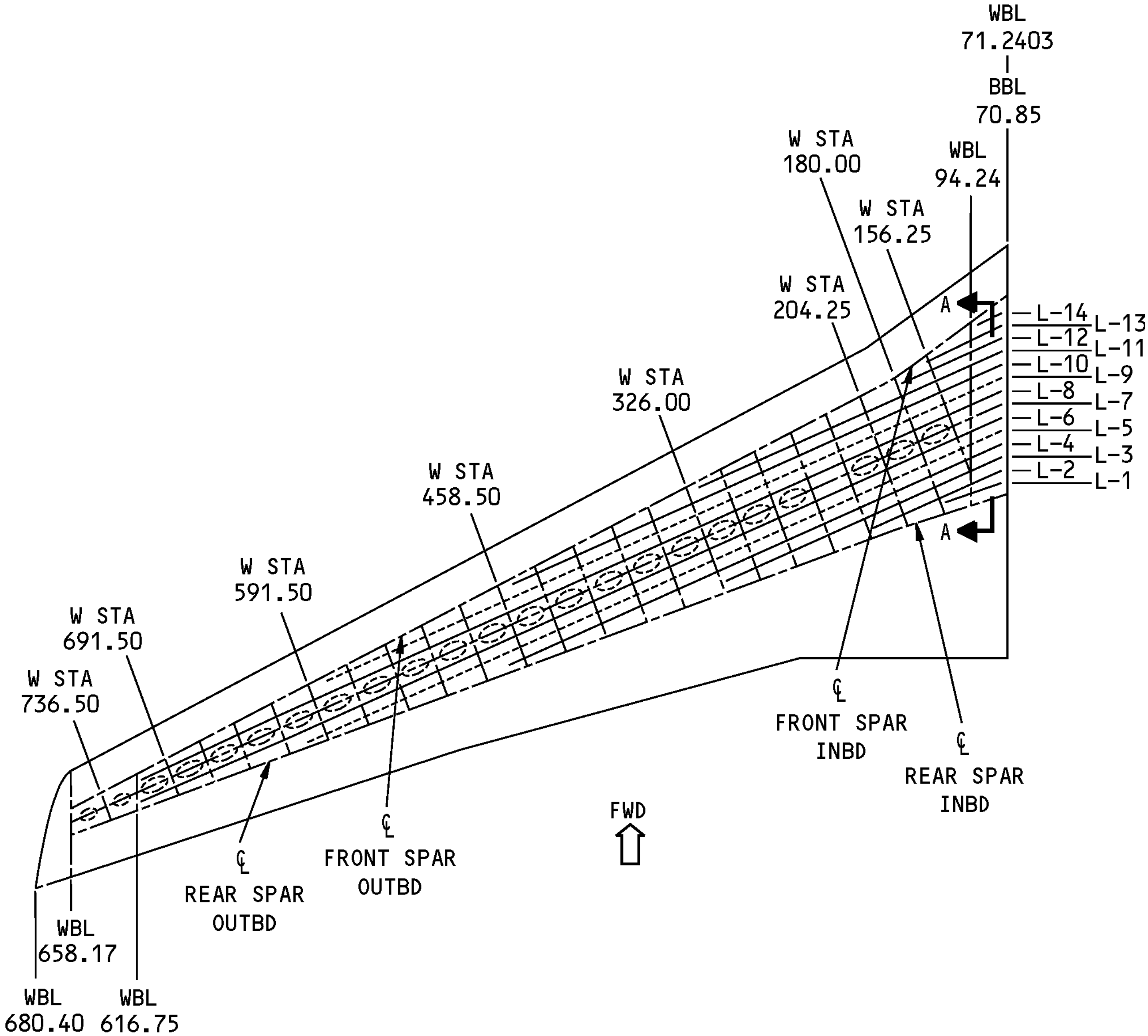
B. Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

C. List of Materials:

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| STRINGER | DESCRIPTION | MATERIAL | DRAWING NUMBER | EFFECTIVITY |
| L-1 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4201 | |
| L-2 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4202 | |
| L-3 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4203 | |
| L-4 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4204 | |
| L-5 | Splice | BAC1506-4261 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-4563) | 112A4205 | |
| L-6 | Zee | BAC1517-2869 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2731) | 112A4206 | For Cum Line 778 and on |
| L-6 | Zee | BAC1517-2731 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2845 or BAC1517-2869) | 112A4206 | For Cum Line 1 thru 777 |
| L-7 | Splice | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4207 | |
| L-8 | Zee | BAC1517-2869 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2845 or BAC1517-2731) | 112A4208 | For Cum Line 778 and on |
| | Zee | BAC1517-2731 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2845 or BAC1517-2869) | 112A4208 | For Cum Line 1 thru 777 |
| L-9 | Splice | BAC1506-4563 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-4261) | 112A4209 | For Cum Line 778 and on |
| | Splice | BAC1506-4261 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-4563) | 112A4209 | For Cum Line 1 thru 777 |
| L-10 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4210 | |
| L-11 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4211 | |

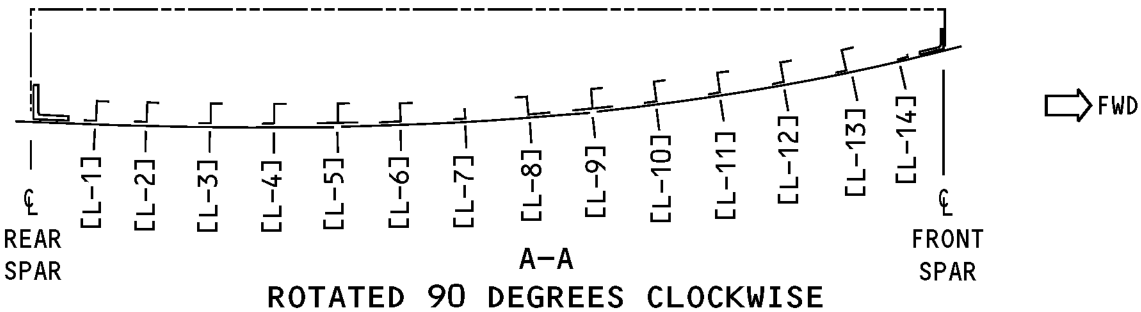
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **STRINGER** | **DESCRIPTION** | **MATERIAL** | **DRAWING NUMBER** | **EFFECTIVITY** |
| L-12 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4212 | |
| L-13 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4213 | |
| L-14 | Zee | BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868) | 112A4214 | |




LEFT WING IS SHOWN,
RIGHT WING IS OPPOSITE

PLAN VIEW OF OUTER WING LOWER SURFACE

**Outer Wing Lower Stringers Identification**
**Figure 2 (Sheet 1 of 2)**




**Outer Wing Lower Stringers Identification**
**Figure 2 (Sheet 2 of 2)**

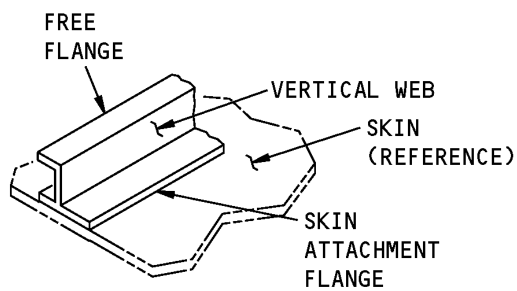
## REPAIR 1 - REPAIR FOR DAMAGE TO AN UPPER SPLICE STRINGER ON THE OUTER WING

### 1. Applicability

A. Repair 1 is applicable to damage to upper splice stringers on the outer wing. A typical upper splice stringer is shown in Figure 201 (Sheet 1).

### 2. General

A. Repair 1 is a Category A damage tolerant repair. Refer to 51-00-06 for the definitions of the different categories of repairs.




**Typical Upper Splice Stringer Section**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| 57-20-01 | OUTER WING SKIN |

(Continued)

| Reference | Title |
|---|---|
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| SOPM 20-10-03 | General - Shot Peening Procedures |

**4. Repair Instructions**

**WARNING:** MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION:** MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 1. Refer to 51-10-02.

**NOTE:** If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 1.

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 201:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Channel | 1 | Use 7075 in the annealed condition that is 0.050 inch thick. Heat treat to the T6 condition after you bend the part |
| [2] | Angle | 1 | Use 7075 in the annealed condition that is 0.050 inch thick. Heat treat to the T6 condition after you bend the part |
| [3] | Channel | 1 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [4] | Angle | 1 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [5] | Angle | 1 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [6] | Plate | 2 | Use 7075-T6 that is 0.190 inch thick |
| [7] | Filler | 1 | Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material |

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-26 or BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-26 or BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.

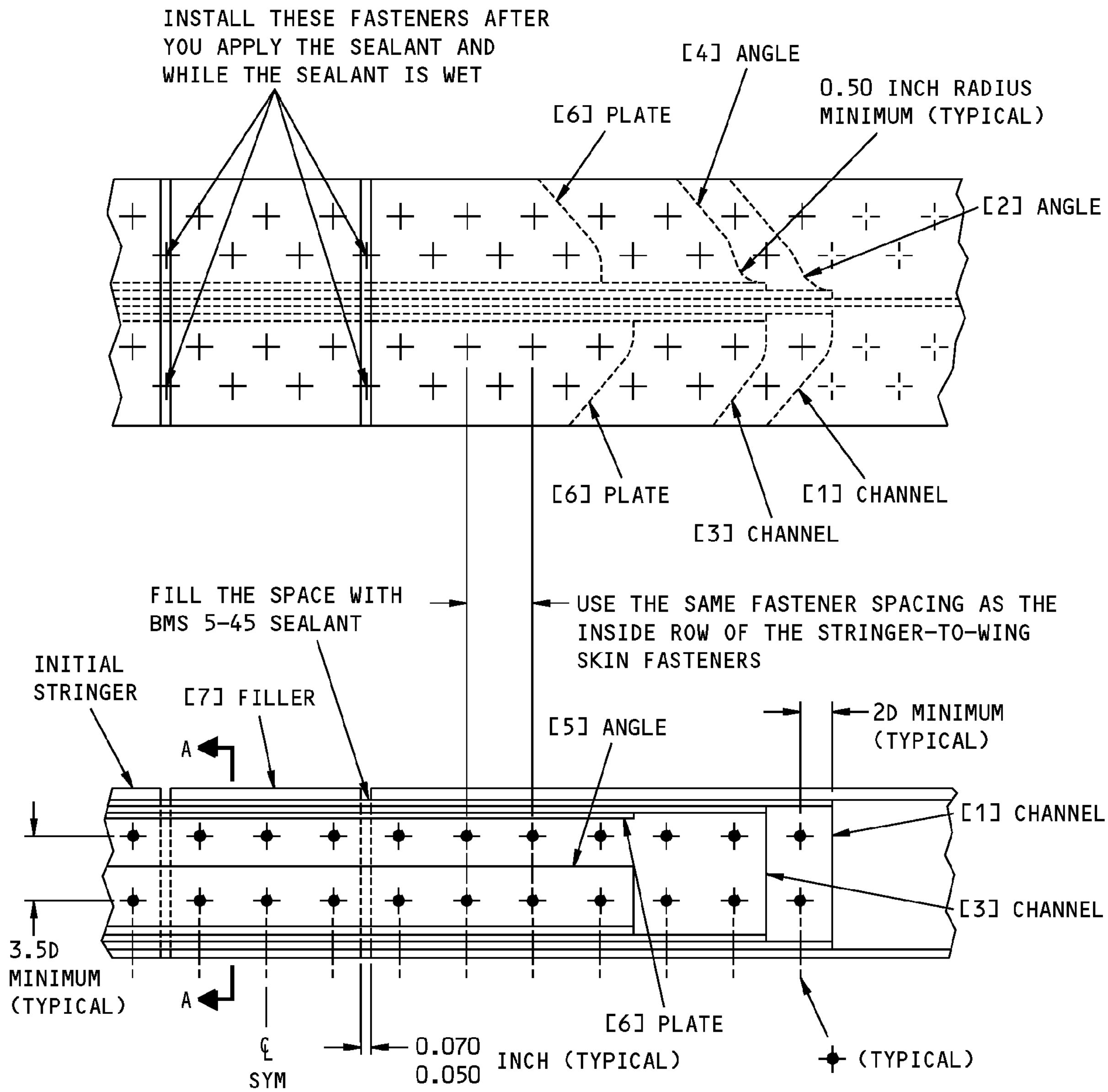
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

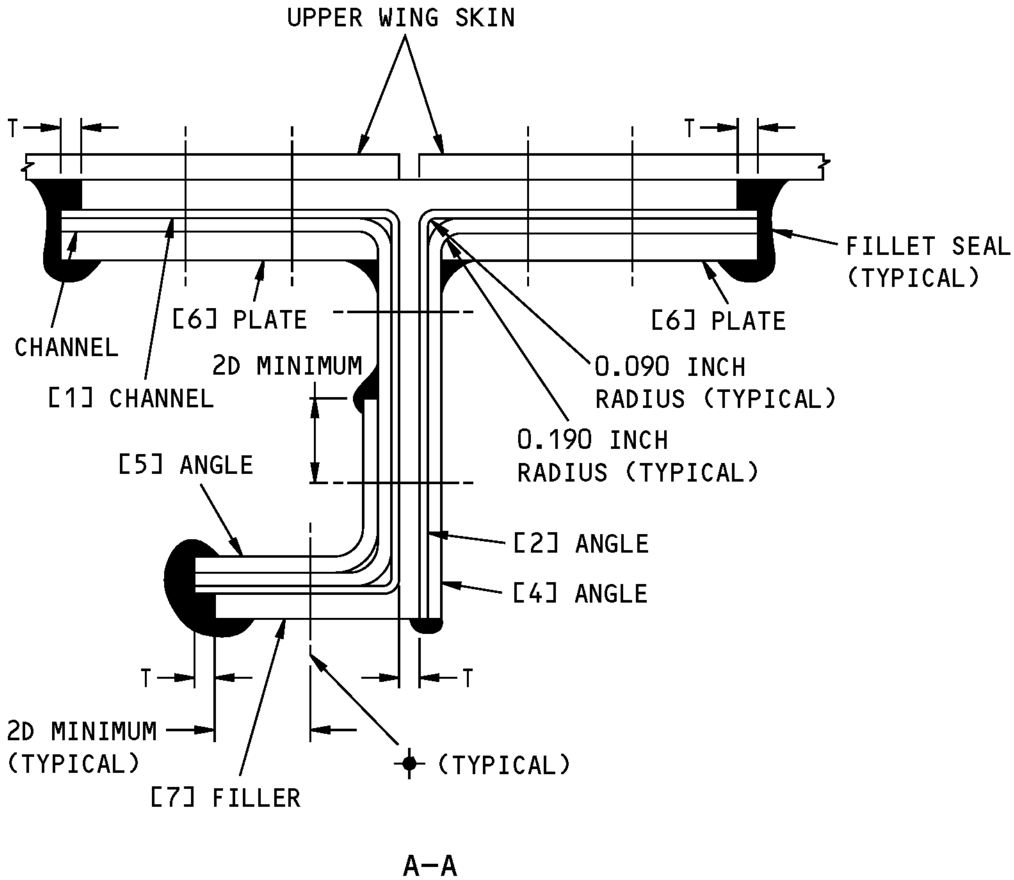
(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.




NOTE: UPPER WING SKIN NOT SHOWN.

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NW()K()Y HEX DRIVE BOLT AND A BACC30R COLLAR.

-◆- REPAIR FASTENER LOCATION. INSTALL A BACB30MY9K() HEX DRIVE BOLT AND A BACC30M COLLAR.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

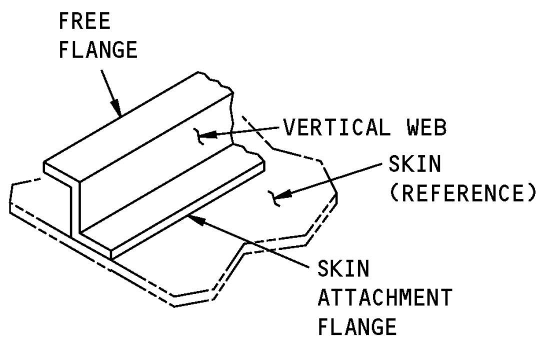
## REPAIR 2 - REPAIR FOR DAMAGE TO AN UPPER ZEE STRINGER ON THE OUTER WING

1. **Applicability**

   A. Repair 2 is applicable to damage to upper zee stringers on the outer wing. A typical upper zee stringer section is shown in Figure 201/REPAIR 2.

2. **General**

   A. Repair 1 is a Category A damage tolerant repair. Refer to 51-00-06 for the definitions of the different categories of repairs.




**Typical Upper Zee Stringer Section**
**Figure 201**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| 57-20-01 | OUTER WING SKIN |

(Continued)

| Reference | Title |
|---|---|
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| SOPM 20-10-03 | General - Shot Peening Procedures |

## 4. Repair Instructions

**WARNING:** MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION:** MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 2. Refer to 51-10-02.

**NOTE**: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 2.

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Angle | 2 | Use 7075 in the annealed condition that is 0.050 inch thick. Heat treat to the T6 condition after you bend the part |
| [2] | Angle | 2 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [3] | Strap | 1 | Use 7075-T6 that is 0.190 inch thick |
| [4] | Filler | 1 | Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material |

F. Assemble the repair parts.
   (1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

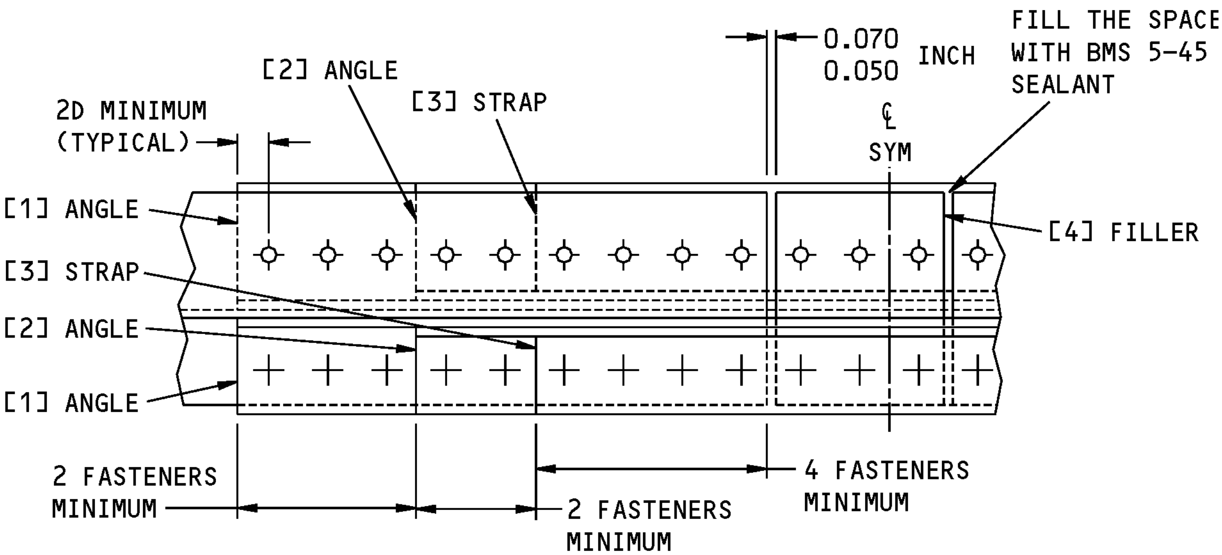
J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

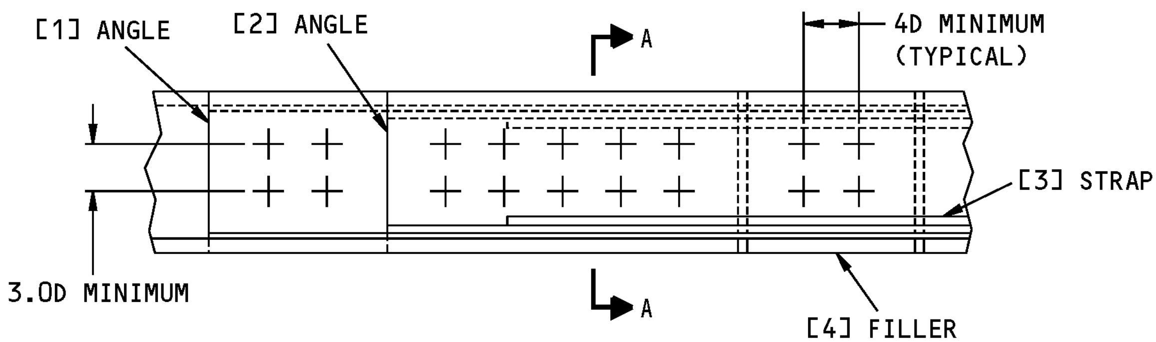
K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.
   (1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.
      (a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.
   (2) Install the repair parts.
      (a) Apply BMS 5-26 or BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.
      (b) Install the fasteners wet with BMS 5-95 sealant. Refer to 51-20-05.
   (3) Fill the space between the part [4] filler and the initial stringer with BMS 5-26 or BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 2.
   (4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.
   (1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.
   (2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

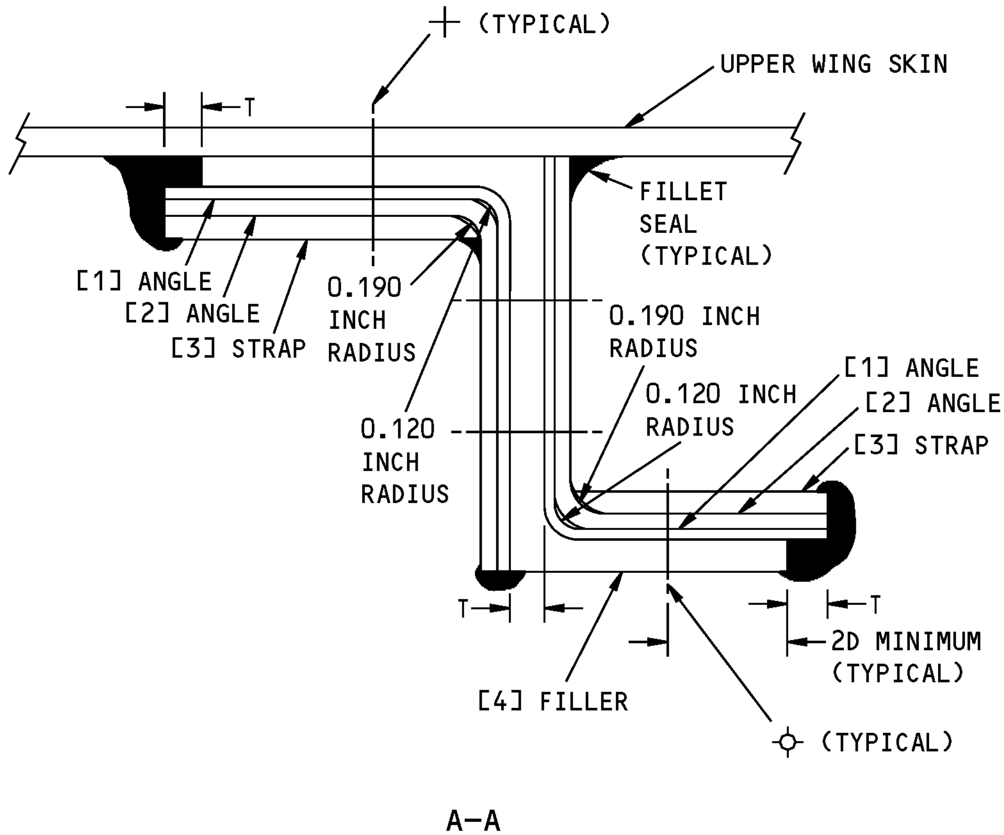
N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.
   (1) Refuel the fuel tank.
   (2) Do a leak test to make sure there are no leaks in the fuel tank.




NOTE: THE UPPER WING SKIN IS NOT SHOWN.

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC COLLAR.

-◆- REPAIR FASTENER LOCATION. INSTALL A BACB30MY10K() HEX DRIVE BOLT AND A BACC30M COLLAR.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

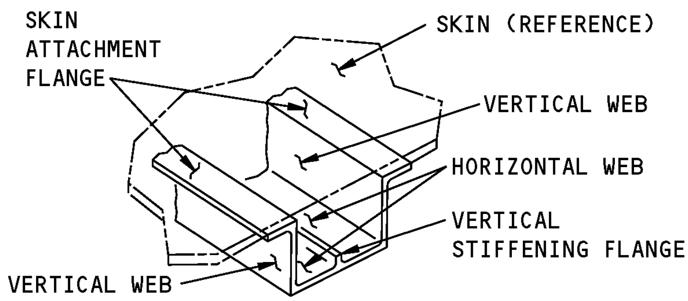
## REPAIR 3 - REPAIR FOR DAMAGE BETWEEN RIBS TO A VENT STRINGER ON THE OUTER WING

### 1. Applicability

A. Repair 3 is applicable to damage between ribs to vent stringers on the outer wing. A typical vent stringer is shown in Figure 201 (Sheet 1).

### 2. General

A. Repair 3 is a Category A damage tolerant repair. Refer to 51-00-06 for the definitions of the different categories of repairs.




**Layout of the Repair Parts**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| 57-20-01 | OUTER WING SKIN |

(Continued)

| Reference | Title |
|---|---|
| 57-20-01, ALLOWABLE DAMAGE 1 | Wing Inspar Skin Inboard of WBL 616.75 |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| SOPM 20-10-03 | General - Shot Peening Procedures |

4. **Repair Instructions**

**WARNING:** MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION:** MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 3. Refer to 51-10-02.

**NOTE**: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 3.

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Angle | As Necessary | Use 7075 in the annealed condition that is as thick as the depth of the recess. Heat treat to the T6 con- dition after you bend the part. If there is not a recess, do not use the part |
| [2] | Angle | 2 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [3] | Angle | 2 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [4] | Plate | 1 | Use 7075-T6 that is 0.050 inch thick |
| [5] | Plate | 2 | Use 7075-T6 that is 0.090 inch thick |
| [6] | Plate | 2 | Use 7075-T6 that is 0.090 inch thick |
| [7] | Filler | 1 | Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material |

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-26 or 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-26 or 5-45 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-26 or BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 3.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.

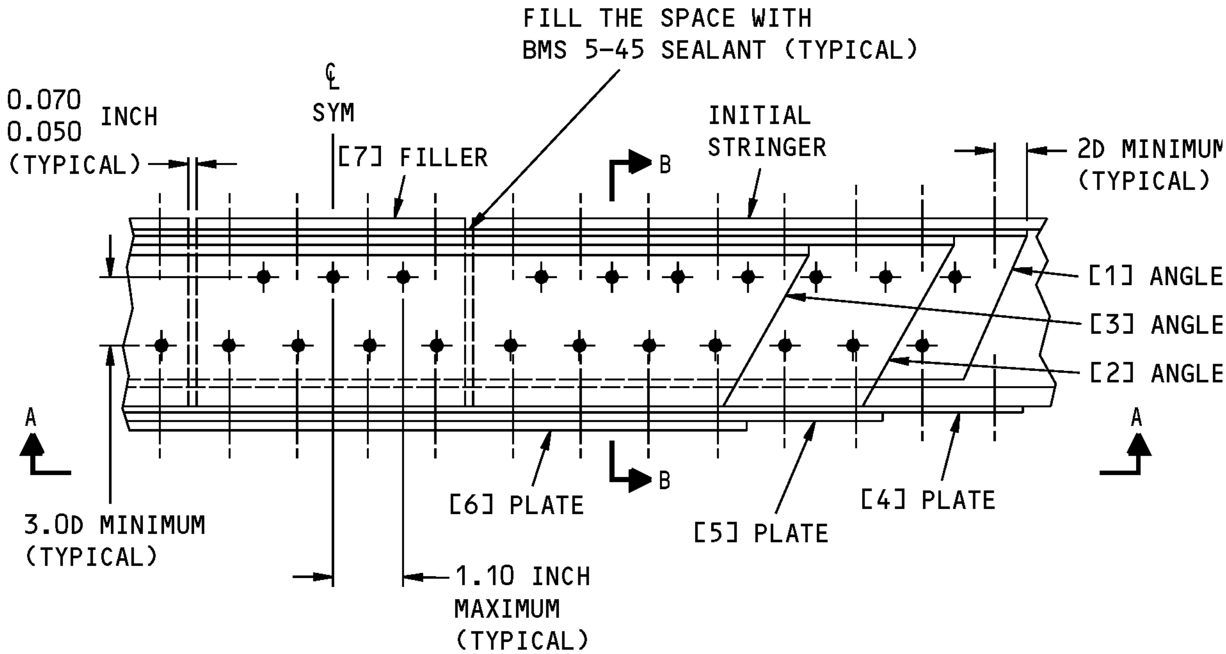
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

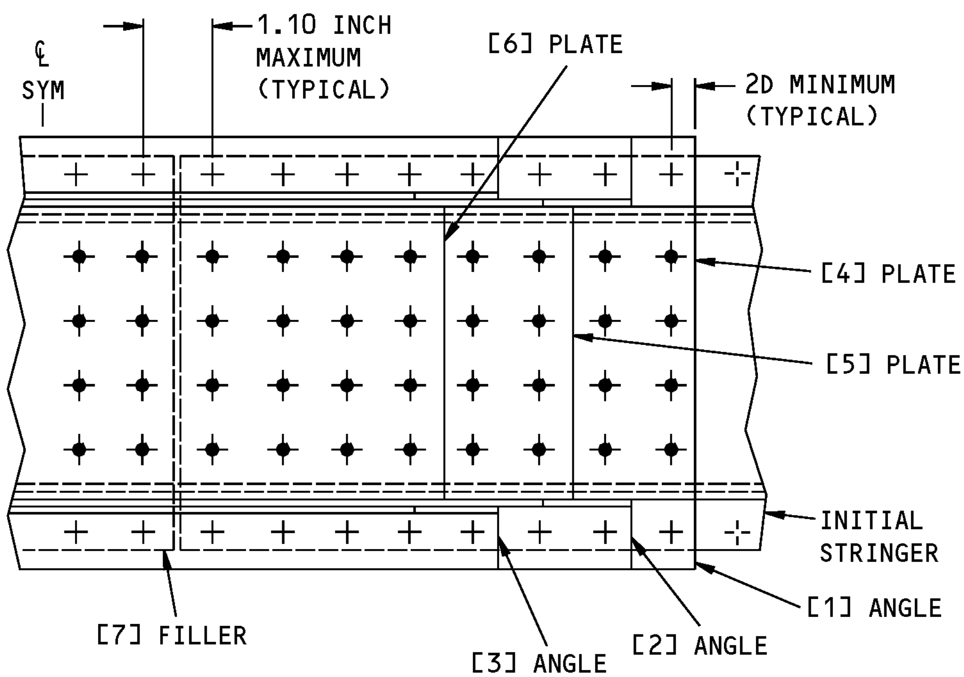
N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

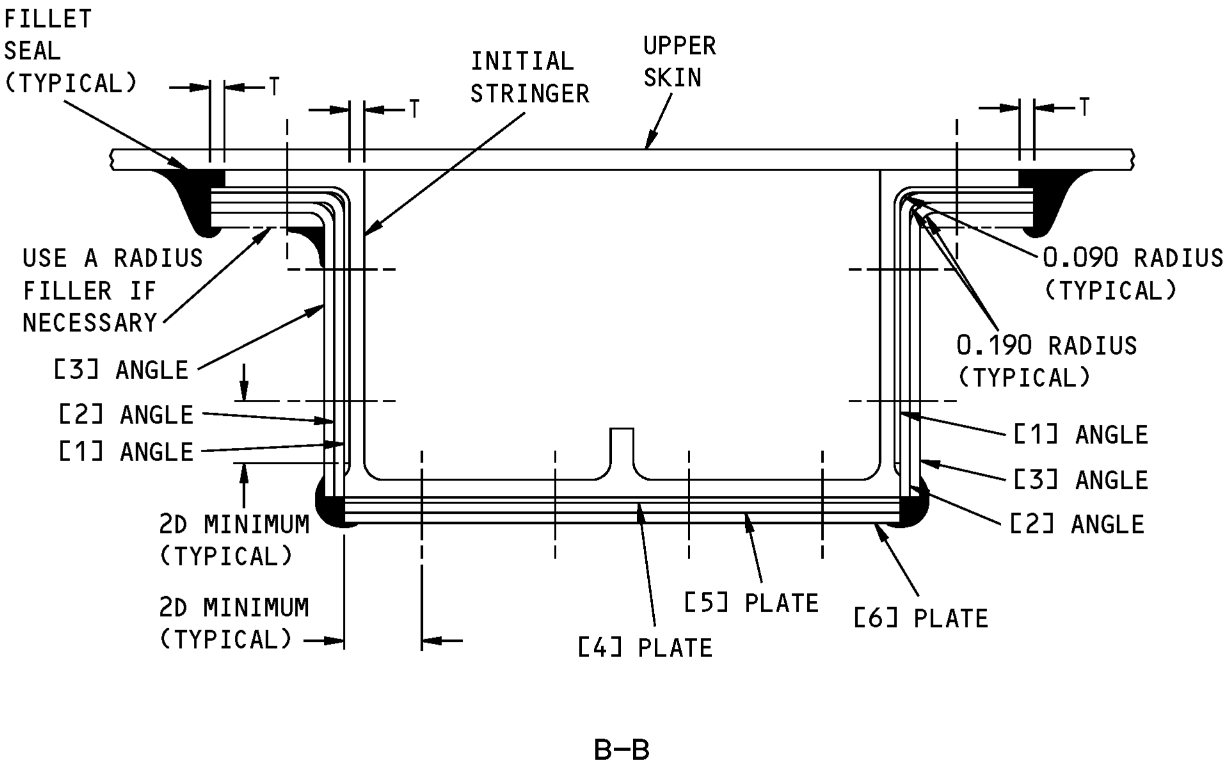



NOTE: UPPER SKIN IS NOT SHOWN.




A-A

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION

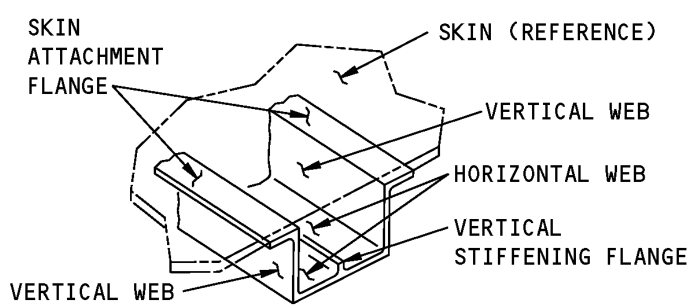
+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K() HEX DRIVE BOLT AND A BACC30AC COLLAR.

⌖ REPAIR FASTENER LOCATION. INSTALL A BACB30VH11CD() BLIND BOLT.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

## REPAIR 4 - REPAIR FOR DAMAGE AT A RIB TO A VENT STRINGER ON THE OUTER WING

1. **Applicability**
   A. Repair 4 is applicable to damage at a rib to vent stringers on the outer wing. A typical vent stringer is shown in Figure 201 (Sheet 1).
2. **General**
   A. Repair 4 is a Category A damage tolerant repair. Refer to 51-00-06 for the definitions of the different categories of repairs.




**Typical Vent Stringer Section**
**Figure 201**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| 57-20-01 | OUTER WING SKIN |

(Continued)

| Reference | Title |
|---|---|
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| SOPM 20-10-03 | General - Shot Peening Procedures |

4. **Repair Instructions**

**WARNING**: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION**: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 4. Refer to 51-10-02.

**NOTE**: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 4.

**WARNING**: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 201:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Angle | As Necessary | Use 7075 in the annealed condition that is as thick as the depth of the recess. Heat treat to the T6 con- dition after you bend the part. If there is not a recess, do not use the part |
| [2] | Angle | 4 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [3] | Angle | 4 | Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part |
| [4] | Filler | As Necessary | Use 7075-T6 that is as thick as the depth of the recess in the stringer. If there is not a recess, do not use the part |
| [5] | Plate | 2 | Use 7075-T6 that is 0.050 inch thick |
| [6] | Plate | 2 | Use 7075-T6 that is 0.090 inch thick |
| [7] | Plate | 2 | Use 7075-T6 that is 0.090 inch thick |
| [8] | Stringer | 1 | Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy |

F. Assemble the repair parts.
   (1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

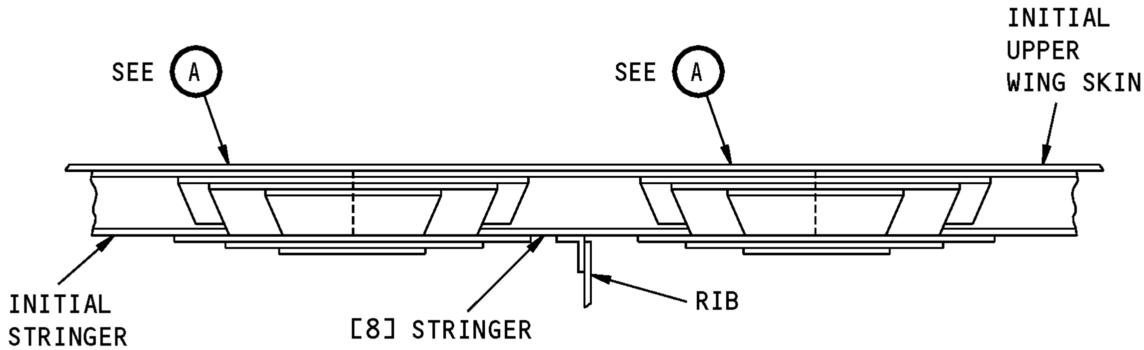
L. Install the repair parts and seal the repair. Refer to 51-20-05.
   (1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.
      (a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.
   (2) Install the repair parts.
      (a) Apply BMS 5-26 or BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.
      (b) Install the fasteners wet with BMS 5-95 sealant. Refer to 51-20-05.
   (3) Fill the space between the part [8] stringer and the initial stringer with BMS 5-26 or BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 4.
   (4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05.

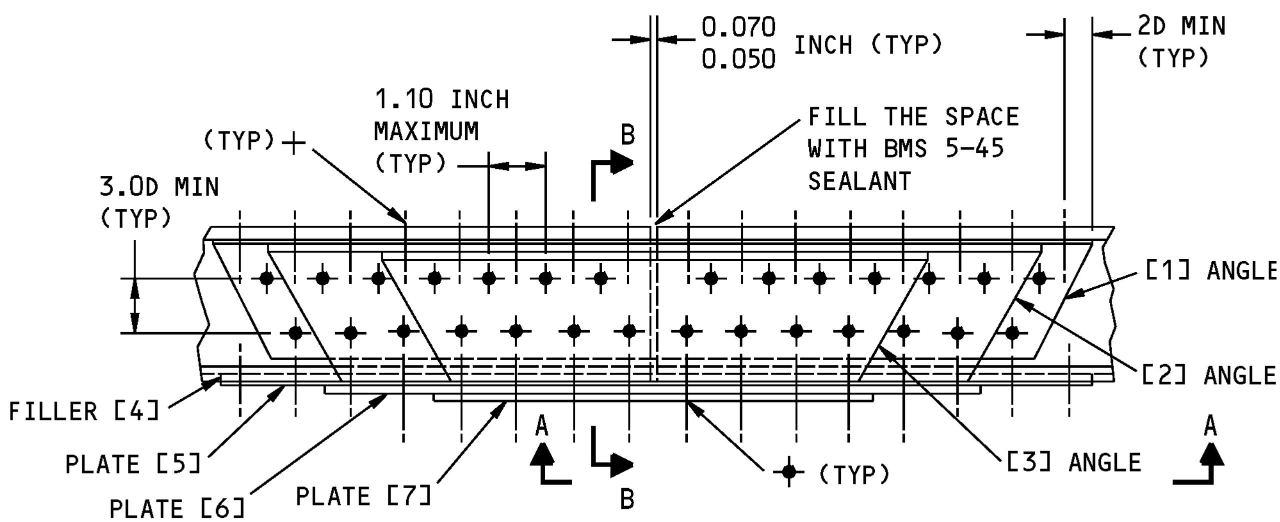
M. Reinstall the fuel tank access doors as necessary.
   (1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.
   (2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

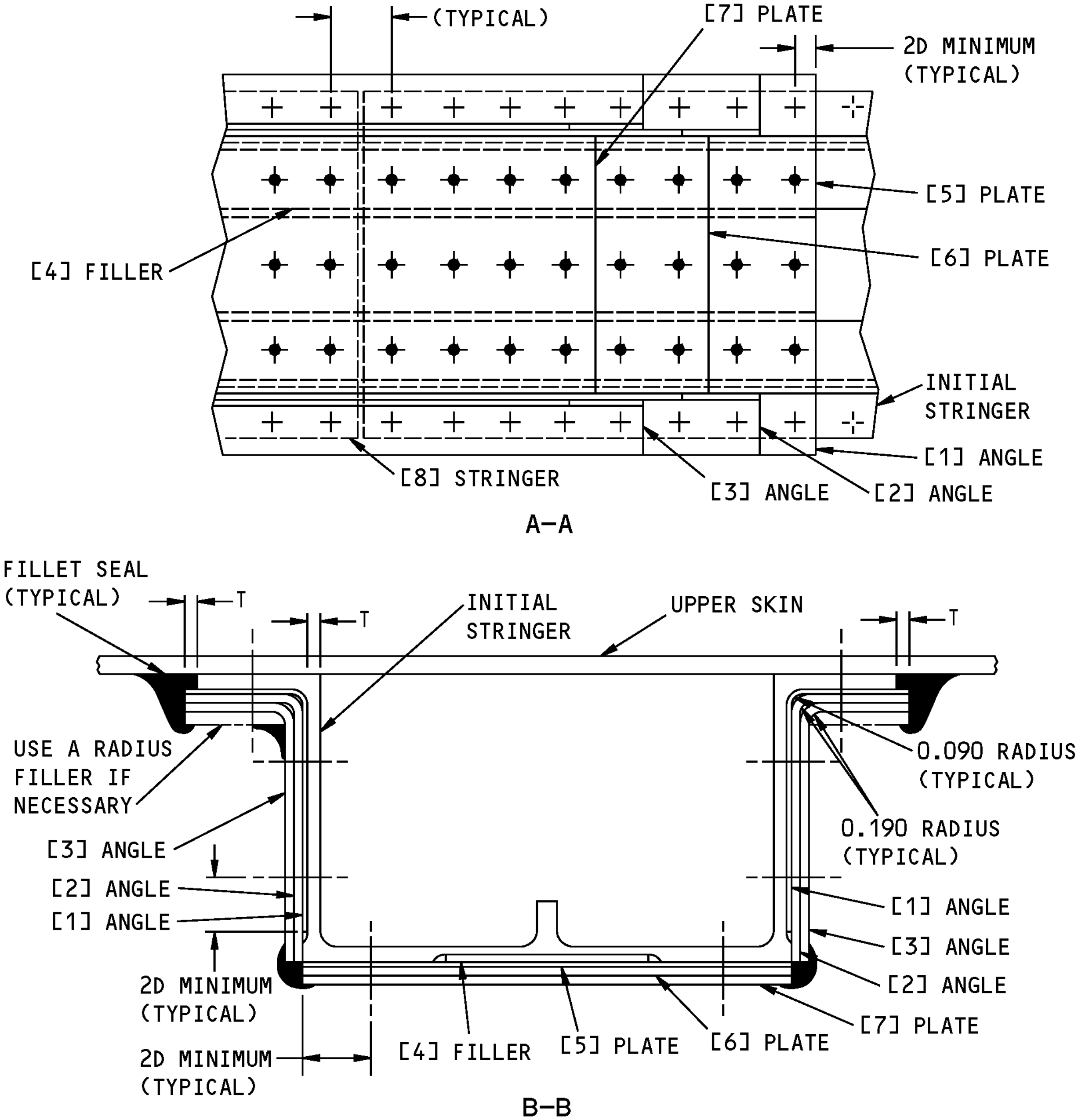



NOTE: INITIAL UPPER WING SKIN NOT SHOWN

A

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION

\+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K() HEX DRIVE BOLT AND A BACC30AC COLLAR.

⊕ REPAIR FASTENER LOCATION. INSTALL A BACB30VH11CD() BLIND BOLT.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

## REPAIR 5 - REPAIR FOR DAMAGE TO A LOWER ZEE STRINGER ON THE OUTER WING

### 1. Applicability

A. Repair 5 is applicable to damage to the lower zee stringers on the outer wing. A typical lower zee stringer is shown in Typical Lower Zee Stringer Section, Figure 201/REPAIR 5.

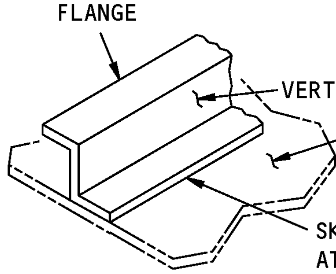
B. Repair 5 is not applicable to damage at a location where the thickness of the stringer is more than 0.250 inch.

### 2. General

A. Repair 5 is a Category A or B repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

(1) Repair to a stringer which is not adjacent to a wing spar or which is not in the region of a fairing is a category A damage tolerant repair

(2) Repair to a stringer which is adjacent to a wing spar or which is in the region of a fairing is a category B damage tolerant repair. For repairs in these areas, contact Boeing.




**Typical Lower Zee Stringer Section**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |

(Continued)

| Reference | Title |
|---|---|
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-50-02 | SUPPORT OF AIRPLANE FOR REPAIR |
| 57-20-01 | OUTER WING SKIN |
| AMM 28-11-00 P/B 201 | FUEL TANKS - MAINTENANCE PRACTICES |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| AMM 28-11-00/801 | Fuel Tanks - Approved Repairs |
| AMM 28-11-11/401 | Wing Fuel Tank Access Panels |
| AMM 28-11-31/401 | Center Fuel Tank Access Panels |
| AMM 28-26-00/201 | Defueling |
| SOPM 20-10-03 | General - Shot Peening Procedures |

## 4. Repair Instructions

**WARNING:** MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

**CAUTION:** MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 5. Refer to 51-10-02.

**NOTE**: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | 1 | Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part |
| [2] | Angle | 1 | Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part |
| [3] | Angle | 1 | Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part |
| [4] | Angle | 1 | Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part |
| [5] | Angle | 1 | Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part |
| [6] | Angle | 1 | Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part |
| [7] | Filler | 1 | Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 2224-T3511 aluminum alloy |
| [8] | Strap | 2 | Use 2024-T3 that is 0.090 inch thick |

E. Make the repair parts. Refer to Table 201/REPAIR 5.

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes.

(1) For repairs made in rib bays 9, 10, and 11, install the initial fasteners in interference fit holes. Refer to Location of Rib Bays 9, 10, and 11, Figure 203/REPAIR 5 for the location of rib bays 9, 10, and 11.

(2) Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-26 or BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

(a) Apply BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-26 or BMS 5-95 sealant. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-26 or BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 5.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05.

M. Reinstall the fuel tank access doors as necessary.

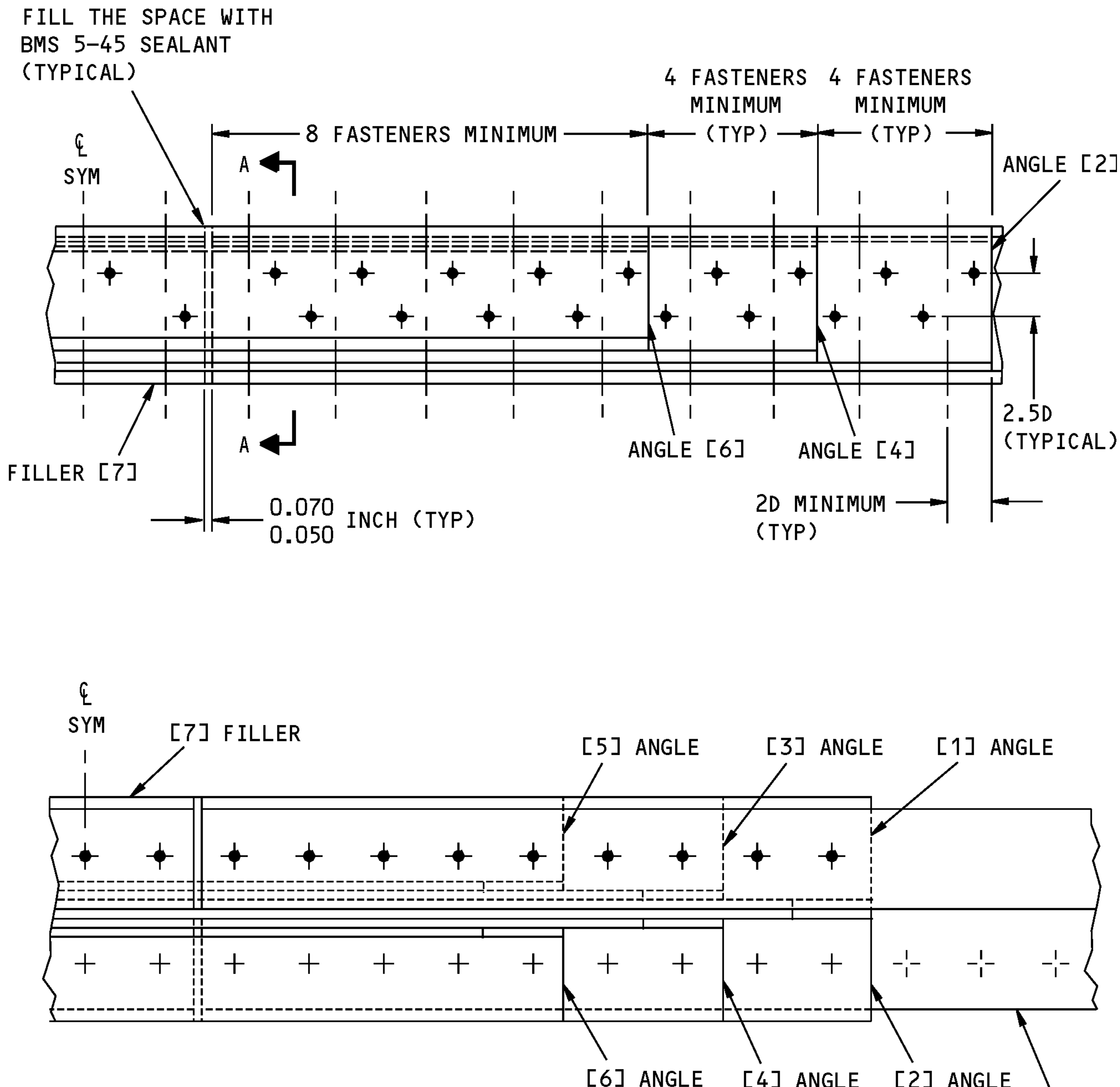
(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

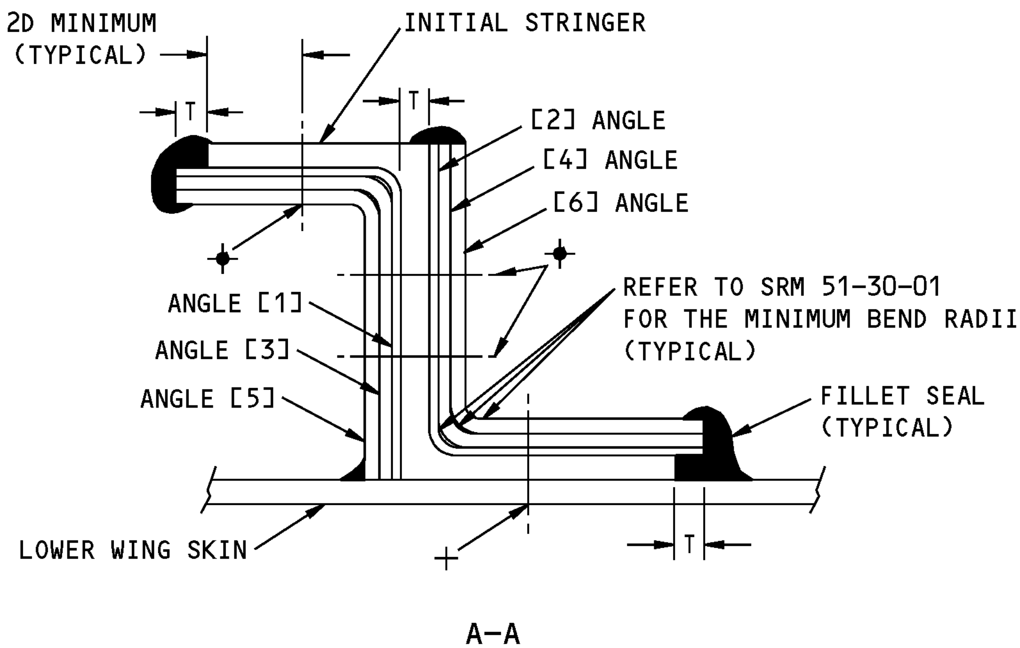
(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.




NOTE: THE LOWER WING SKIN IS NOT SHOWN.

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**



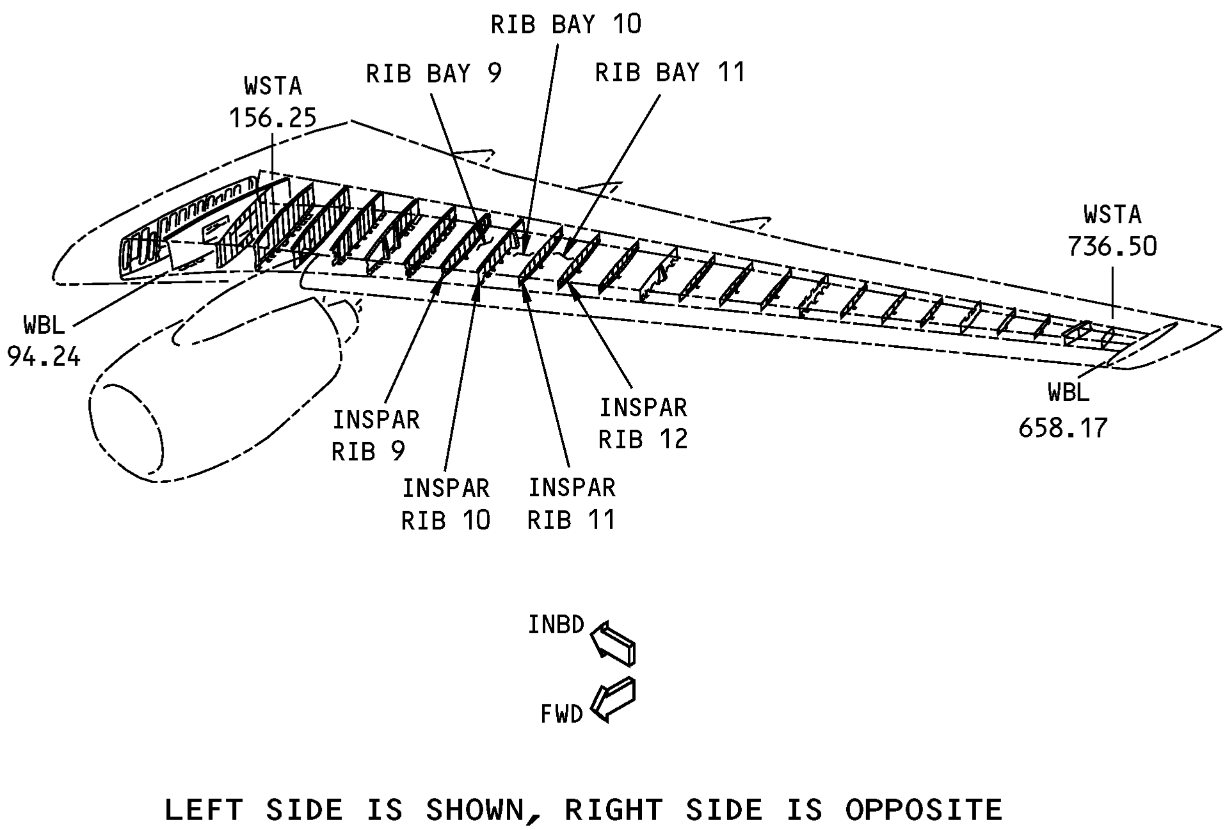

FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION

\+ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC COLLAR. IN RIB BAYS 9, 10, AND 11 INSTALL THE FASTENER IN AN INTERFERENCE FIT HOLE. REFER TO FIGURE 203 FOR THE LOCATION OF RIB BAYS 9, 10, AND 11.

-◆- REPAIR FASTENER LOCATION. INSTALL A BACB30MY10K() HEX DRIVE BOLT.

**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**



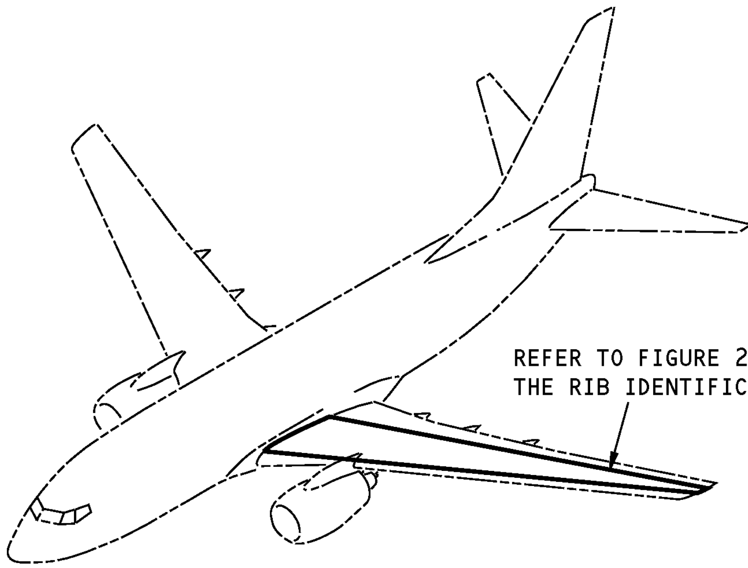

**Location of Rib Bays 9, 10, and 11**
**Figure 203**

5. **Inspection Instructions**

A. Ask the Boeing Company for the inspection instructions.

## IDENTIFICATION 1 - OUTER WING RIBS

### 1. Locator




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Outer Wing Rib Locations**
**Figure 1**

### 2. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 112A0015 | Functional Product Collector - Inspar Ribs and Wing Tip, Left Side |
| 112A0016 | Functional Product Collector - Inspar Ribs and Wing Tip, Right Side |
| 112A5010 | Side-Of-Body Installation, BBL 70.85 |
| 112A5020 | Rib Installation - Inspar Rib 2, WBL 94.24 |
| 112A5030 | Rib Installation - Inspar Rib 3, WSTA 156.25 |
| 112A5040 | Rib Installation - Inspar Rib 4, WSTA 180.00 |
| 112A5050 | Rib Installation - Inspar Rib 5, WSTA 204.25 |
| 112A5060 | Rib Installation - Inspar Rib 6, WSTA 228.25 |
| 112A5070 | Rib Installation - Inspar Rib 7, WSTA 253.00 |
| 112A5080 | Rib Installation - Inspar Rib 8, WSTA 278.50 |
| 112A5090 | Rib Installation - Inspar Rib 9, WSTA 302.50 |
| 112A5100 | Rib Installation - Inspar Rib 10, WSTA 326.00 |
| 112A5110 | Rib Installation - Inspar Rib 11, WSTA 353.00 |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 112A5120 | Rib Installation - Inspar Rib 12, WSTA 378.50 |
| 112A5130 | Rib Installation - Inspar Rib 13, WSTA 405.50 |
| 112A5140 | Rib Installation - Inspar Rib 14, WSTA 432.50 |
| 112A5150 | Rib Installation - Inspar Rib 15, WSTA 458.50 |
| 112A5160 | Rib Installation - Inspar Rib 16, WSTA 485.00 |
| 112A5170 | Rib Installation - Inspar Rib 17, WSTA 512.00 |
| 112A5180 | Rib Installation - Inspar Rib 18, WSTA 537.50 |
| 112A5190 | Rib Installation - Inspar Rib 19, WSTA 564.50 |
| 112A5200 | Rib Installation - Inspar Rib 20, WSTA 591.50 |
| 112A5210 | Rib Installation - Inspar Rib 21, WSTA 617.00 |
| 112A5220 | Rib Installation - Inspar Rib 22, WSTA 643.50 |
| 112A5230 | Rib Installation - Inspar Rib 23, WSTA 667.50 |
| 112A5240 | Rib Installation - Inspar Rib 24, WSTA 691.50 |
| 112A5250 | Rib Installation - Inspar Rib 25, WBL 616.75 |
| 112A5260 | Rib Installation - Inspar Rib 26, WSTA 736.50 |
| 112A5270 | Rib Installation - Closure Rib 27, WBL 658.17 |

## 3. Applicability

A. Identification 1 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 1 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

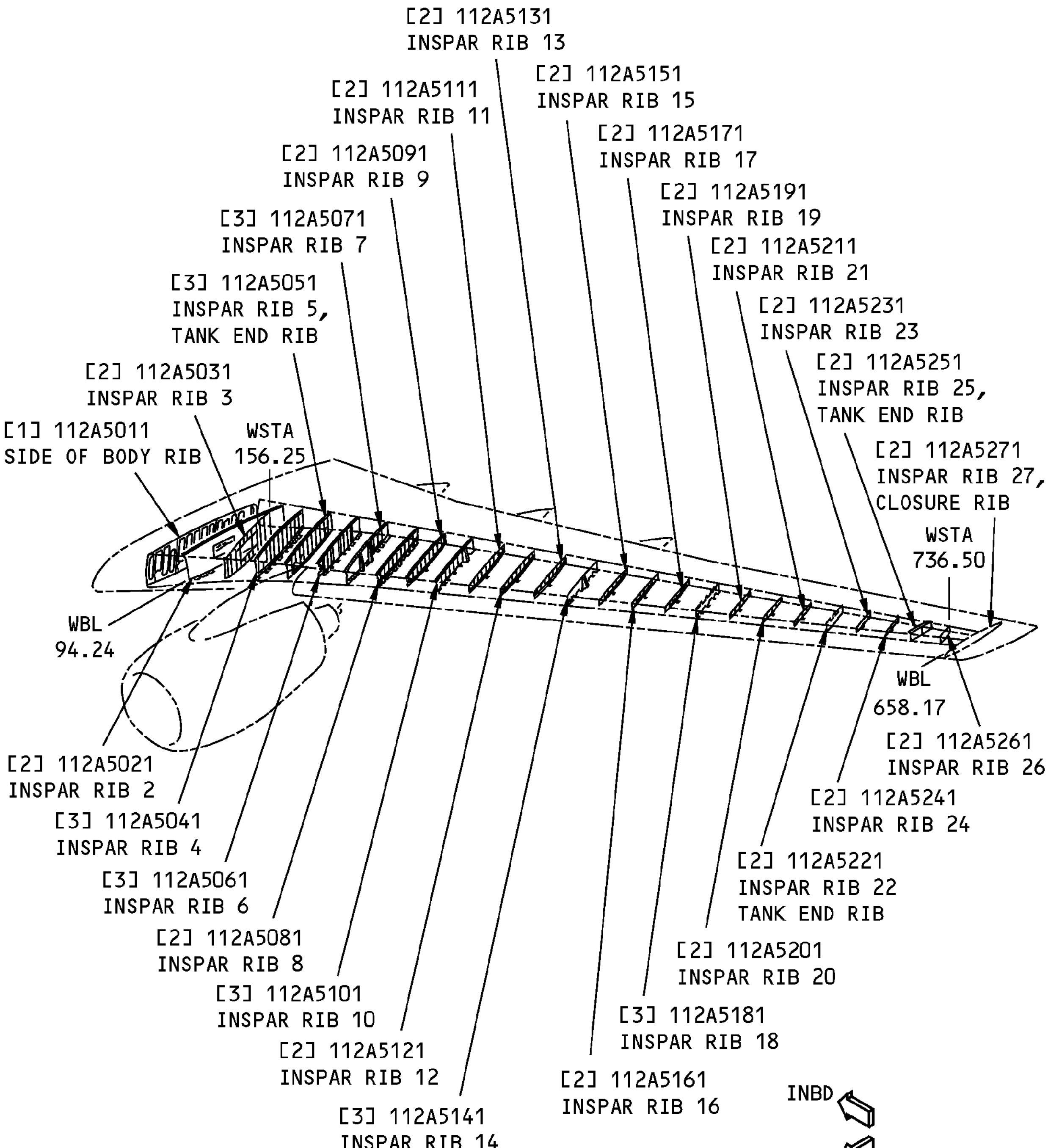
B. List of Materials:

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Side of Body Rib Assembly | | Refer to Figure 3 | |
| | Web, Machined | | 7150-T651 aluminum plate as given in BMS 7-256. Refer to Figure 3 for the machined thicknesses | |
| | Forward Web | 0.500 (12.7) | | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | Mid Web | 0.500 (12.7) | | |
| | Aft Web | 0.700 (17.78) | | |
| | Stiffener 1 (8) | | BAC1520-2798 7150-T77511 aluminum extrusion as given in BMS 7-306 | |
| | Stiffener 2 (3) | | BAC1506-4454 7150-T77511 aluminum extrusion as given in BMS 7-306 | |
| | Stiffener 3 (2) | | BAC1505-100900 7150-T77511 aluminum extrusion as given in BMS 7-306 | |
| | Stiffener 4 | | BAC1520-2797 7150-T77511 aluminum extrusion as given in BMS 7-306 | |
| | Upper Rib Chord | | BAC1520-2775 7075-T73 aluminum extrusion as given in QQ-A-200/11 | |
| | Lower Rib Chord | | BAC1520-2754 2024-T42 aluminum extrusion as given in QQ-A-200/3 | |
| [2] | Rib, Machined | | 7050-T7451 aluminum plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses. Refer to Figure 4 for an example of a typical machined rib. | |
| | | | 7050-T7451 aluminum plate as given in BMS 7-323. Refer to the production drawing for the machined thicknesses. Refer to Figure 4 for an example of a typical machined rib. | Cum Line 789 and on with winglet full provision installation. |
| [3] | Rib, Machined | | 7050-T7451 aluminum plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined thicknesses. Refer to Figure 5 for an example of a typical machined rib. | |

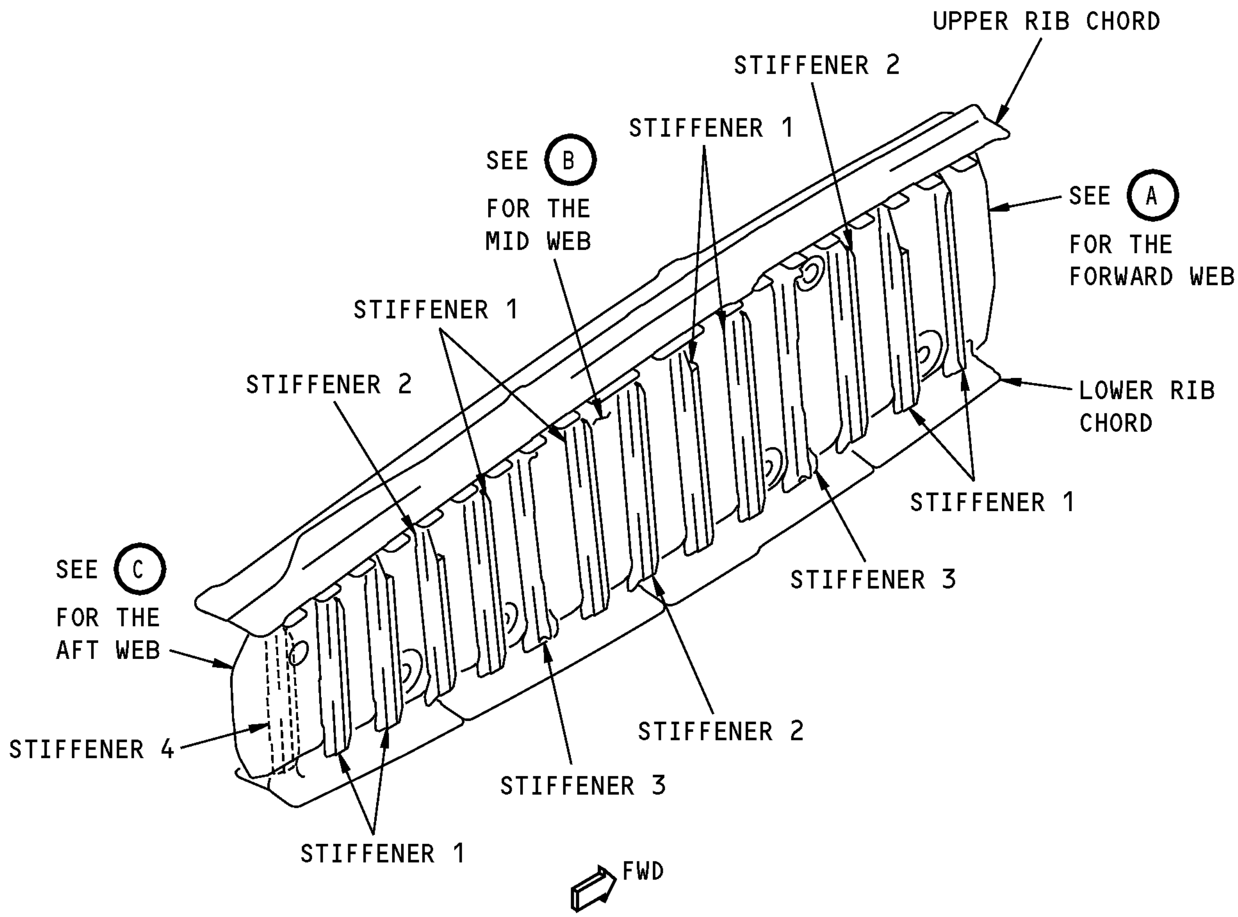
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

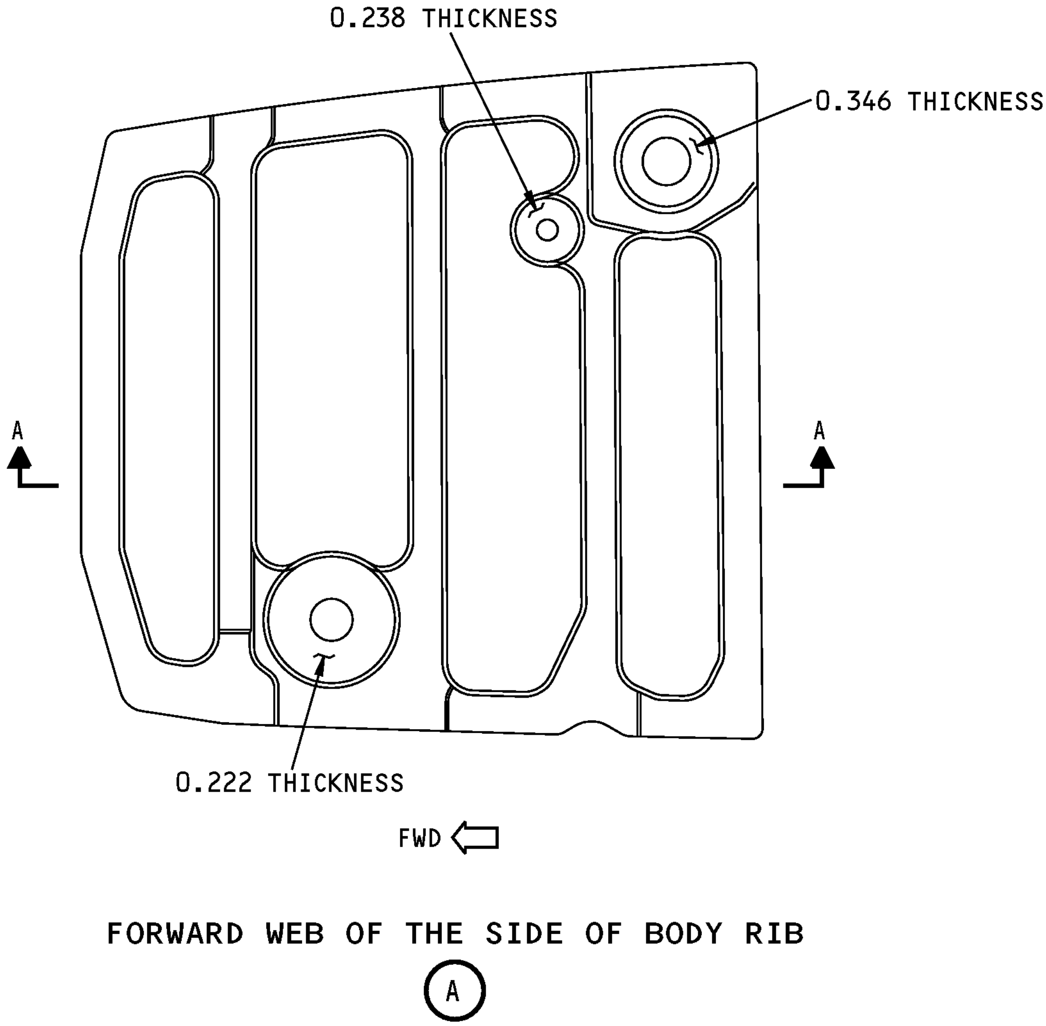
**Outer Wing Rib Identification**
**Figure 2**

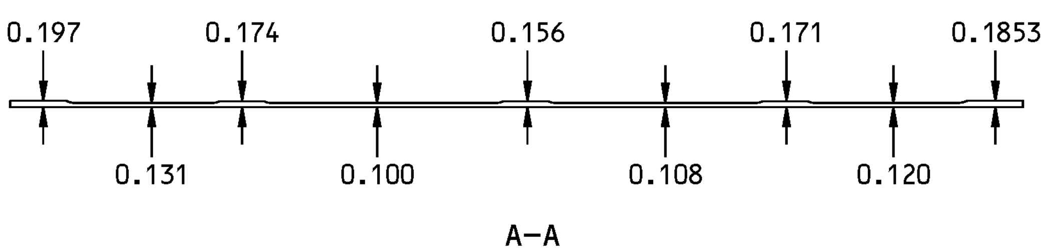



NOTES

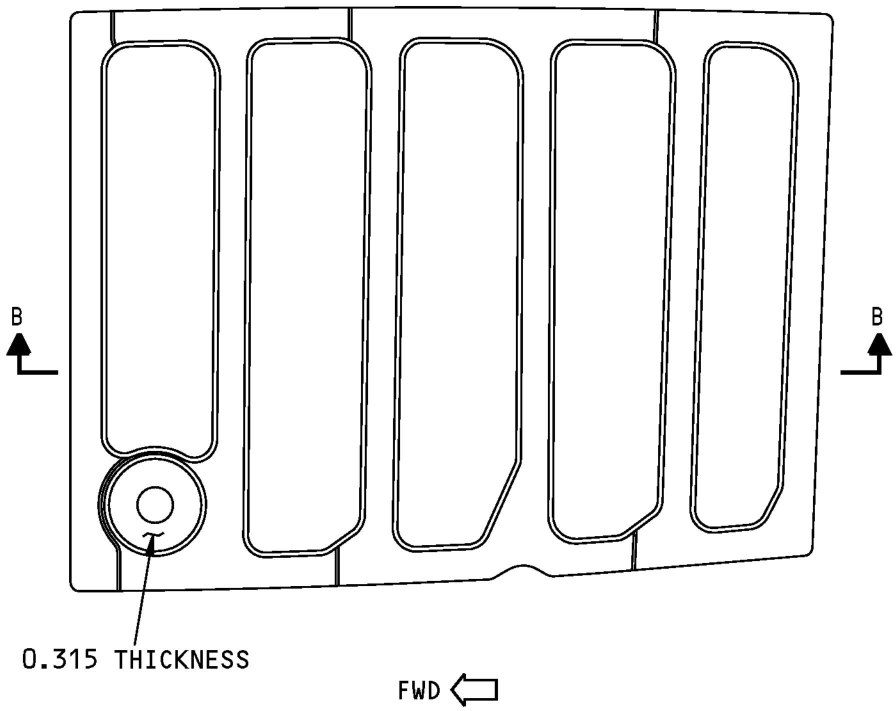
- ALL DIMENSIONS ARE IN INCHES.

**Side of Body Rib Assembly**
**Figure 3 (Sheet 1 of 4)**



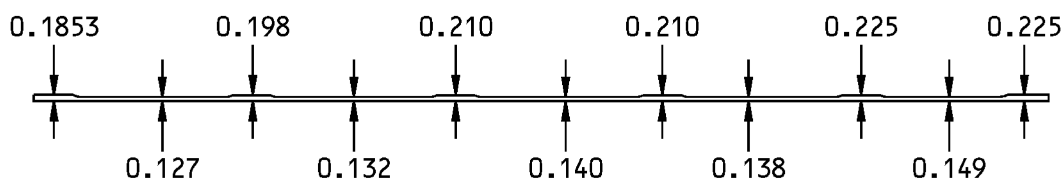




**Side of Body Rib Assembly**
**Figure 3 (Sheet 2 of 4)**



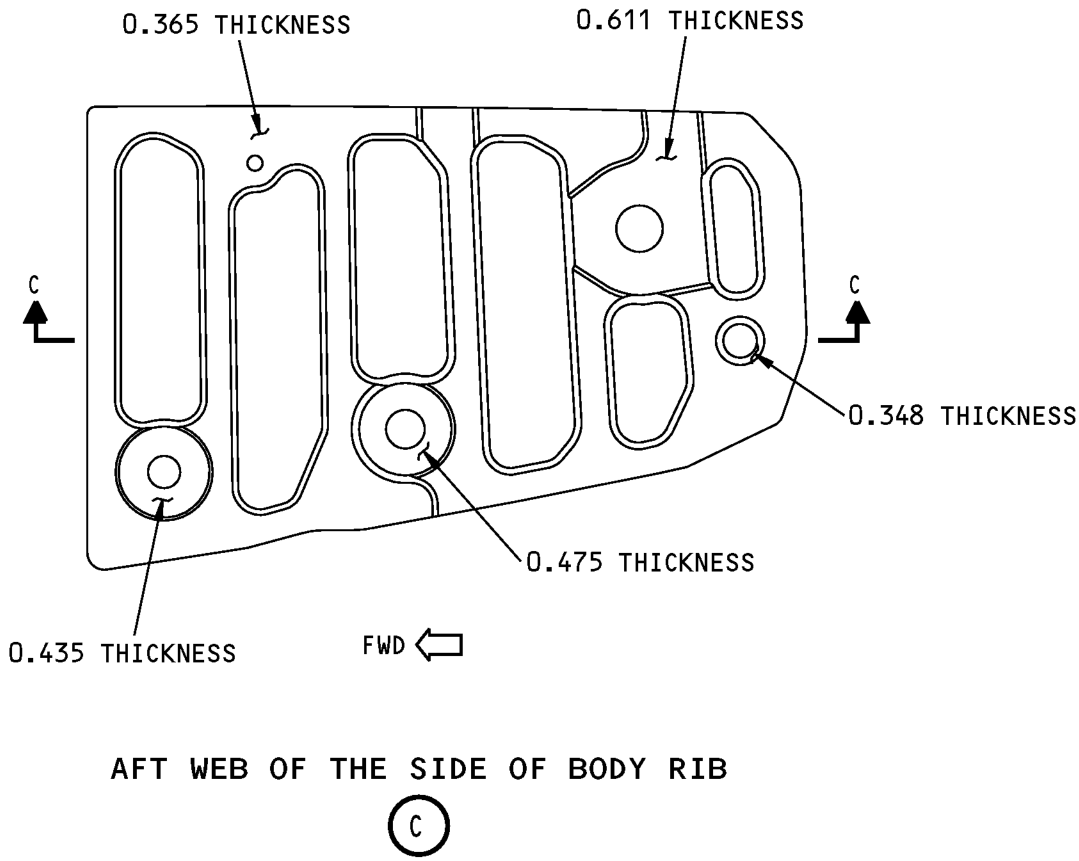

MID WEB OF THE SIDE OF BODY RIB

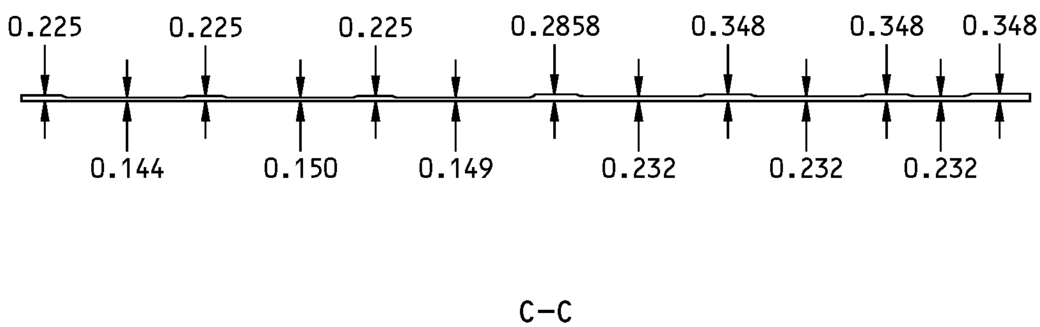
B



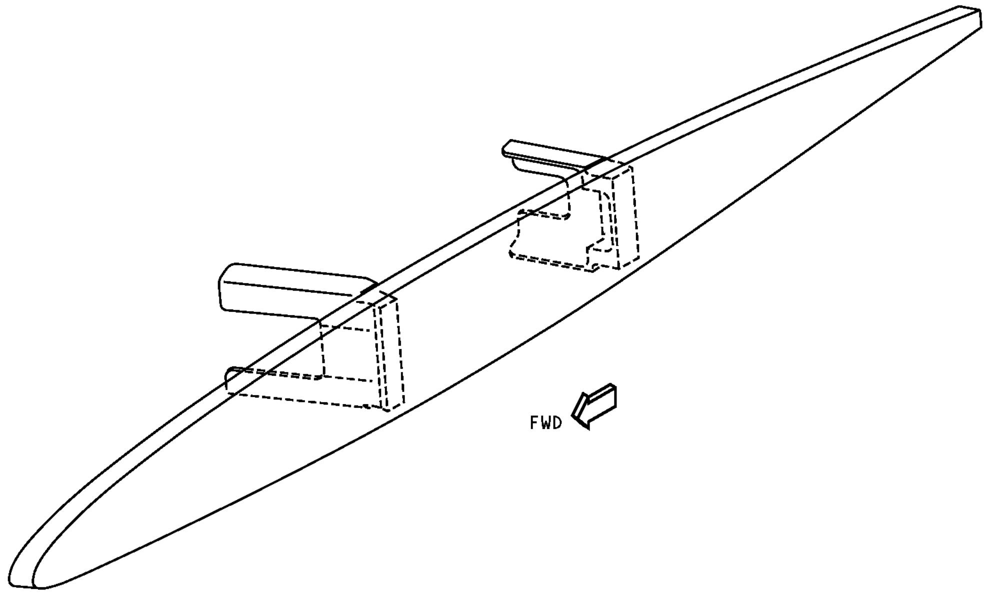

B-B

**Side of Body Rib Assembly**
**Figure 3 (Sheet 3 of 4)**

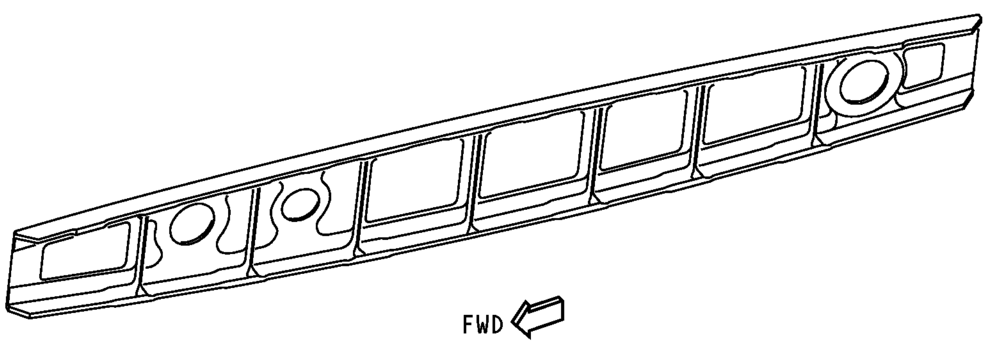



**Side of Body Rib Assembly**
**Figure 3 (Sheet 4 of 4)**

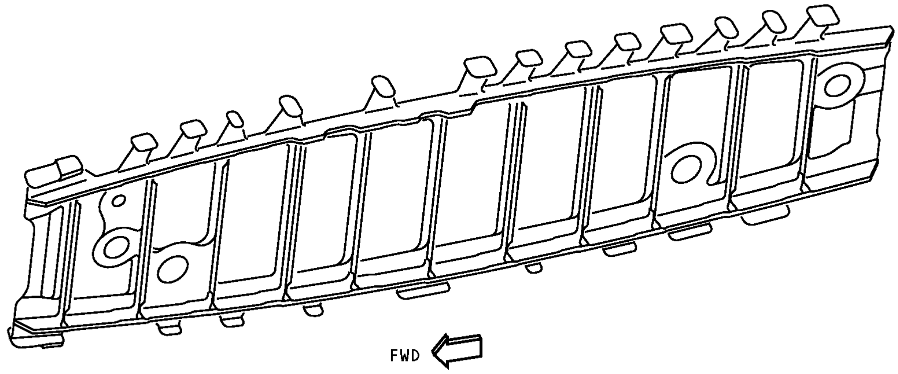



MACHINED RIB
(FOR CUM LINE NUMBER 789 AND ON
WITH FULL WINGLET PROVISION INSTALLATION)




MACHINED RIB (TYPICAL)

**Typical Outer Wing Machined Rib for Figure 2, Item [2]**
**Figure 4**

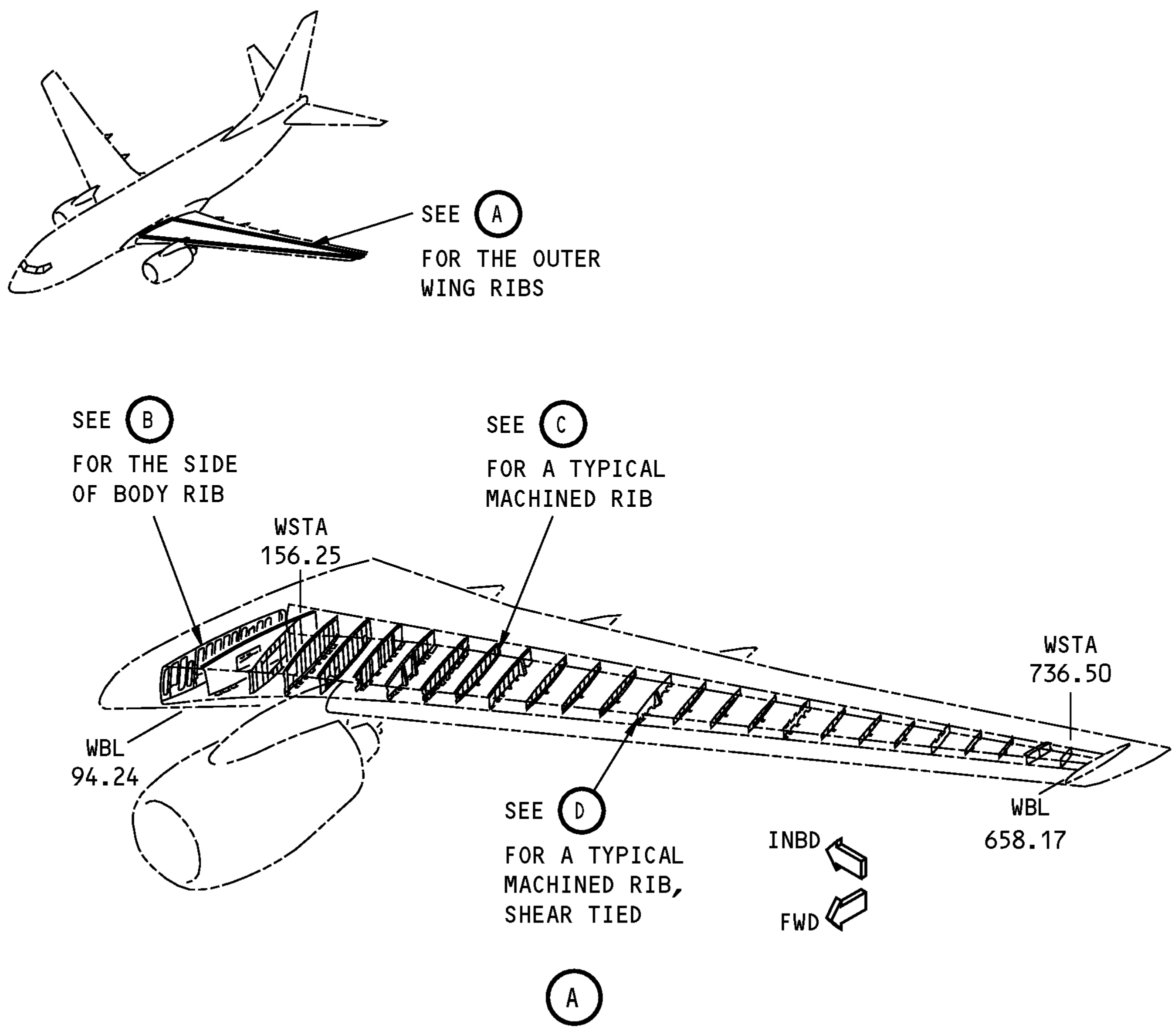



**Typical Outer Wing Machined Rib for Figure 2, Item [3]
Figure 5**

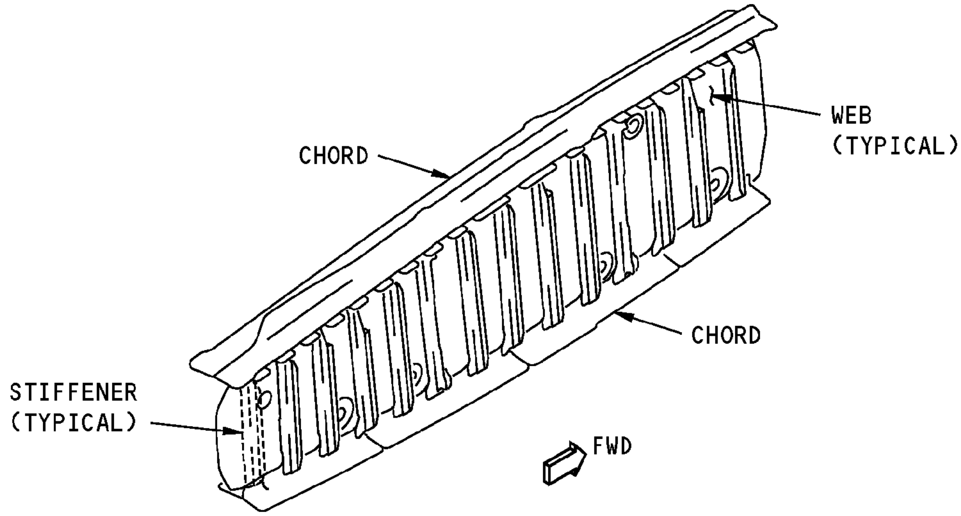
## ALLOWABLE DAMAGE 1 - OUTER WING RIBS

### 1. Applicability

A. This subject gives the allowable damage limits for the outer wing ribs shown in Outer Wing Rib Locations, Figure 101/ALLOWABLE DAMAGE 1.

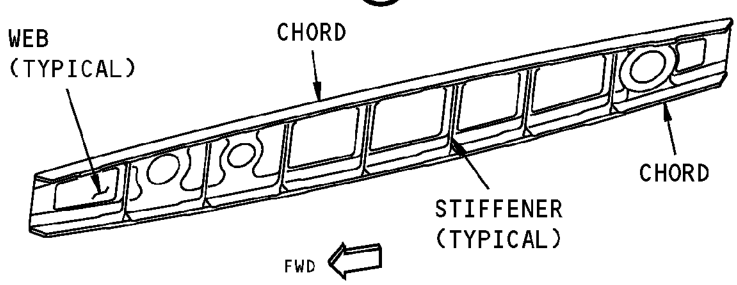



**Outer Wing Rib Locations**
**Figure 101 (Sheet 1 of 2)**



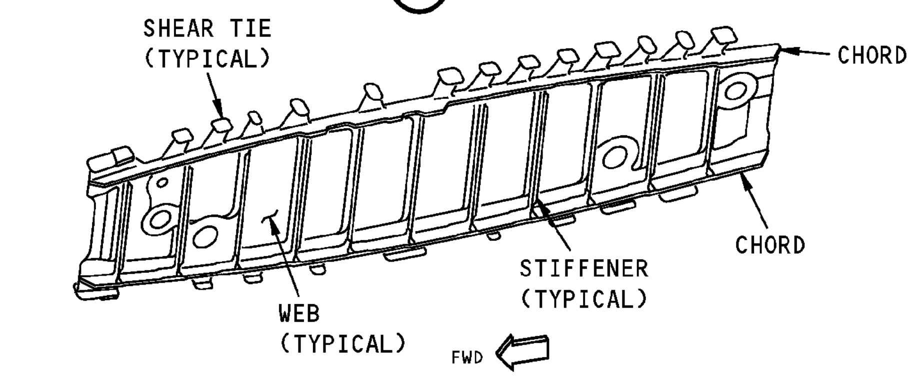

SIDE OF BODY RIB (ALUMINUM)

B




TYPICAL MACHINED RIB (ALUMINUM)

C




TYPICAL MACHINED RIB, SHEAR TIED (ALUMINUM)

D

**Outer Wing Rib Locations**
**Figure 101 (Sheet 2 of 2)**

## 2. General

**WARNING:** MAKE SURE THAT ALL THE FUEL IS REMOVED FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. FUEL VAPORS CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Remove the parts as necessary to get access to the damaged area.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

C. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of abrasives and other materials you can use to remove damage.

(3) Refer to 51-30-05 for possible sources of equipment and tools you can use to remove damage.

D. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(2) Apply a layer of BMS 10-20, Type II primer to the surface of the reworked area. Refer to SOPM 20-41-02.

(3) Apply BMS 5-26 or BMS 5-45 sealant to damaged fillet seals as given in 51-20-05.

(a) Refer to 51-20-05 for the sealing of tank end ribs.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Side of Body Rib

(1) Web

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A and B .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , D , and G .

(c) Dents:

1) Refer to Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail E for the damage that is permitted.

(d) Holes and Punctures are not permitted.

(2) Chord

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and F .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , D , and G .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(3) Stiffener

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and F .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail A , B , C , D , F , and G .

(c) Dents are not permitted:

(d) Holes and Punctures are not permitted.

B. Inspar Ribs 2 through 27

(1) Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail A , B , C , D , and G .

(c) Dents:

1) Refer to Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail E for the damage that is permitted.

(d) Holes and Punctures are not permitted.

(2) Chord

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and F .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , D , F , and G .

(c) Dents:

1) Refer to Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail E for the damage that is permitted.

(d) Holes and Punctures are not permitted.

(3) Stiffener

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and F .

(b) Nicks, Gouges, Scratches, and Corrosion:

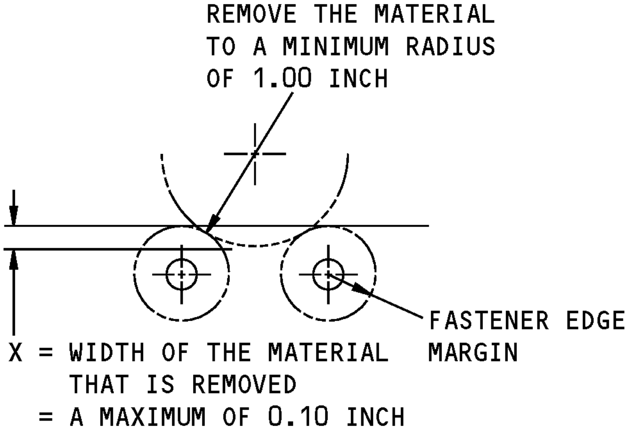
1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , D , F , and G .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

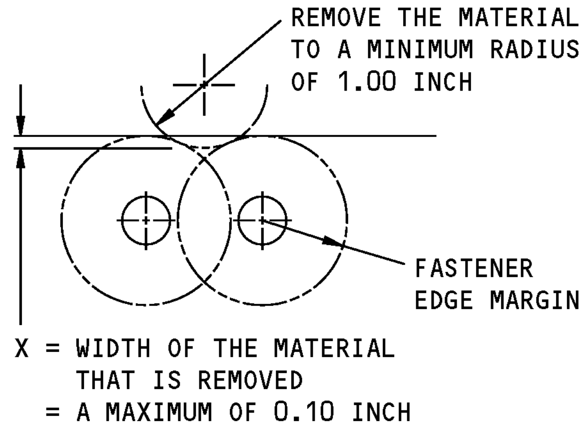
(4) Shear tie

(a) Damage is not permitted.



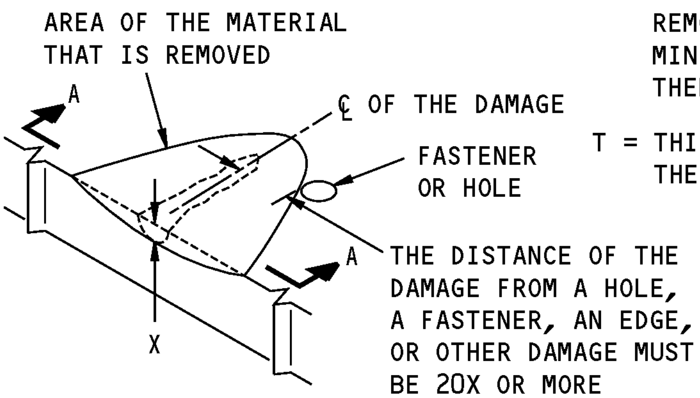

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

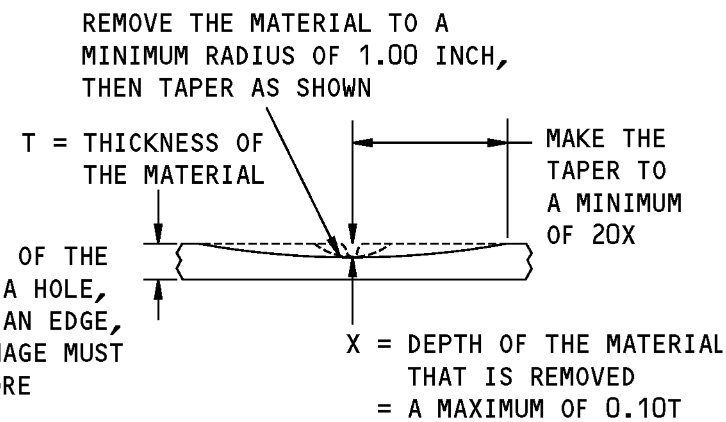



NOTE: THE MAXIMUM CROSS-SECTIONAL AREA OF MATERIAL THAT IS REMOVED MUST NOT BE MORE THAN 4% OF THE INITIAL CROSS-SECTIONAL AREA.

FOR FINDING THE CROSS-SECTIONAL AREA, SEE G.

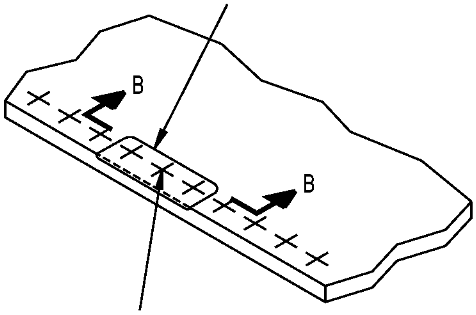
REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C



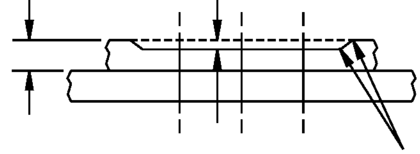

A-A

**Allowable Damage Limits**
**Figure 102 (Sheet 1 of 4)**

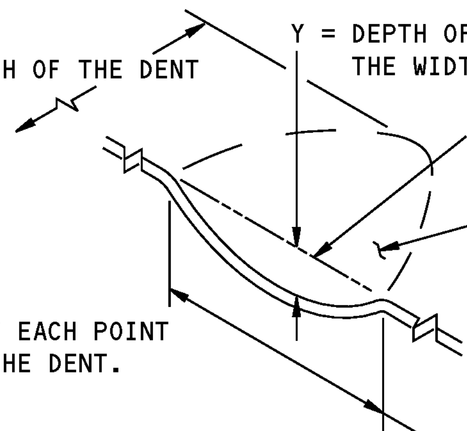



REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

D




B-B




$\frac{W}{Y}$ MUST BE 30 OR MORE AT EACH POINT ALONG THE LENGTH OF THE DENT.

SIDE OF BODY RIB WEB:
Y = A MAXIMUM OF 0.05 INCH FOR THE WEB.

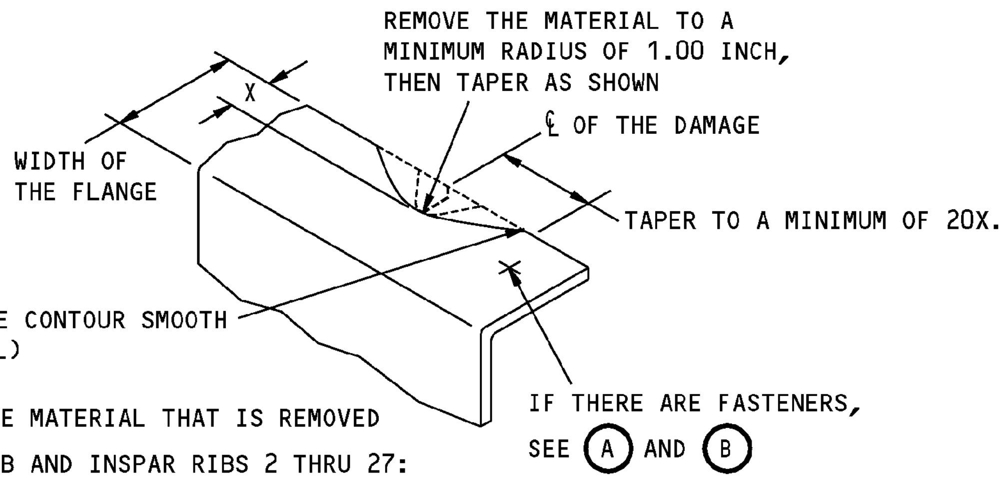
INSPAR RIBS 2 THRU 27 CHORD:
Y = A MAXIMUM OF 0.125 INCH.

INSPAR RIBS 2 THRU 27 WEB:
Y = A MAXIMUM OF 0.05 INCH

DENT THAT IS PERMITTED

E

**Allowable Damage Limits**
**Figure 102 (Sheet 2 of 4)**

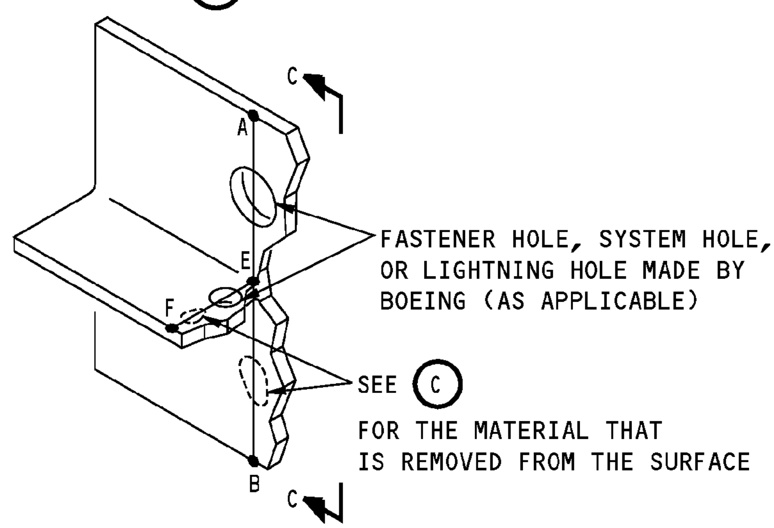



X = WIDTH OF THE MATERIAL THAT IS REMOVED

SIDE OF BODY RIB AND INSPAR RIBS 2 THRU 27:
X = A MAXIMUM OF 0.10 INCH.

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

F



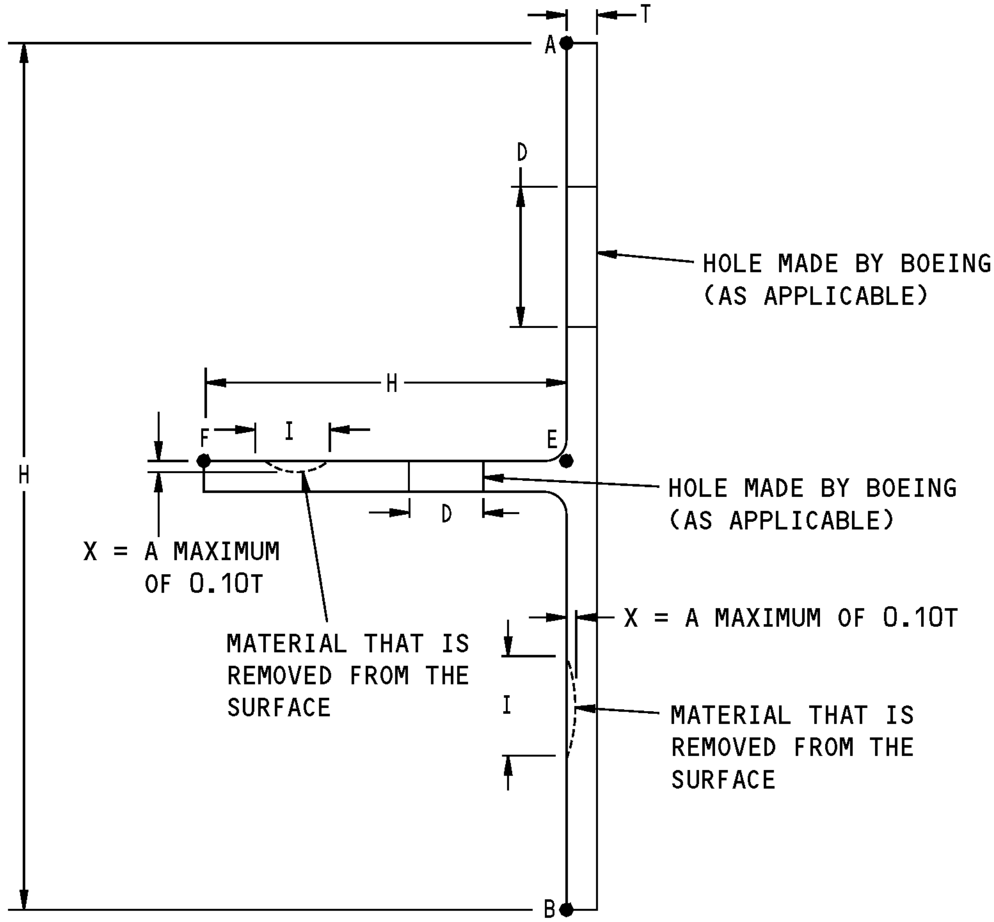

NOTE: THE CROSS-SECTIONAL AREA OF THE DAMAGE WHICH HAS BEEN REMOVED ALONG ANY LINE A-B OR E-F MUST NOT BE MORE THAN 4 PERCENT OF THE INITIAL CROSS-SECTIONAL AREA.

IF THERE ARE NO INITIAL HOLES MADE BY BOEING, DO NOT USE "D" WHEN FINDING THE INITIAL CROSS-SECTIONAL AREA.

CROSS-SECTIONAL AREA REMOVAL

G

**Allowable Damage Limits**
**Figure 102 (Sheet 3 of 4)**



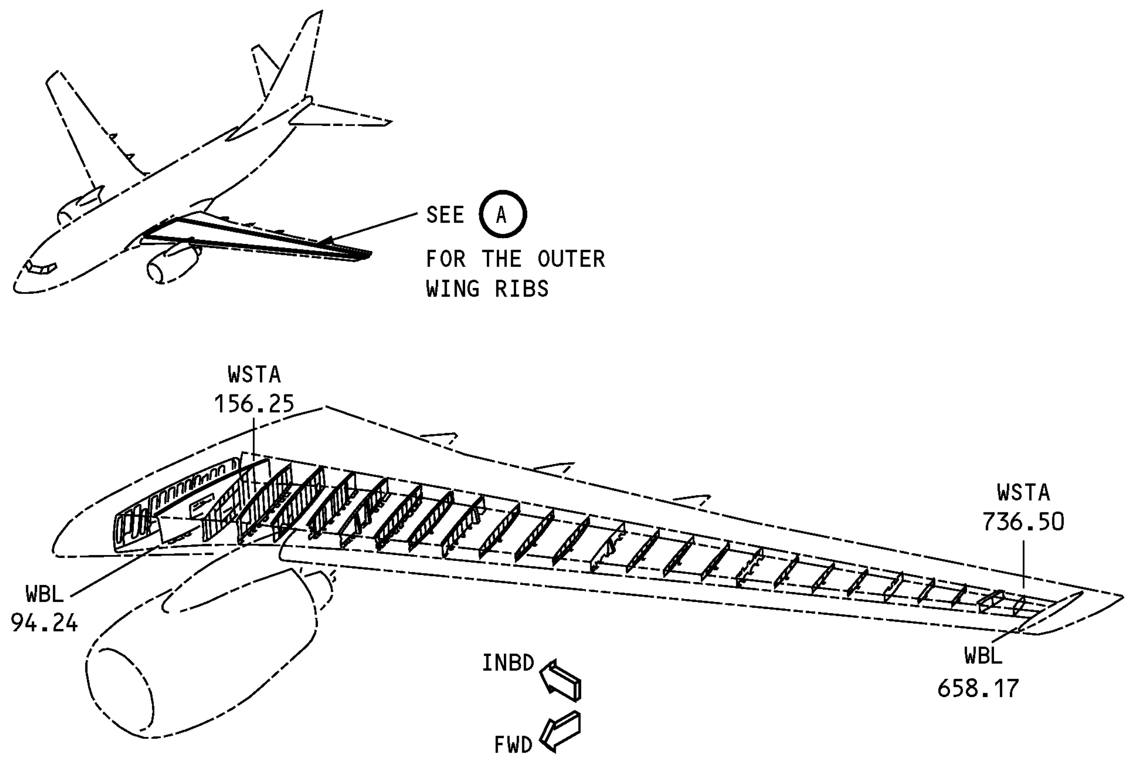

EXAMPLE: $A_1$ = INITIAL CROSS-SECTIONAL AREA
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE INITIAL HOLE CROSS-SECTIONAL AREA MADE BY BOEING.
= TH-TD

$A_2$ = CROSS-SECTIONAL AREA OF THE DAMAGE WHICH HAS BEEN REMOVED
= THE DEPTH OF THE DAMAGE REMOVED MULTIPLIED BY THE DIMENSION ACROSS THE DAMAGE REMOVED
= XI

$\left(\frac{A_2}{A_1}\right) \leq$ 4% MAXIMUM

**Allowable Damage Limits**
**Figure 102 (Sheet 4 of 4)**

## REPAIR 1 - OUTER WING RIB REPAIRS




NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

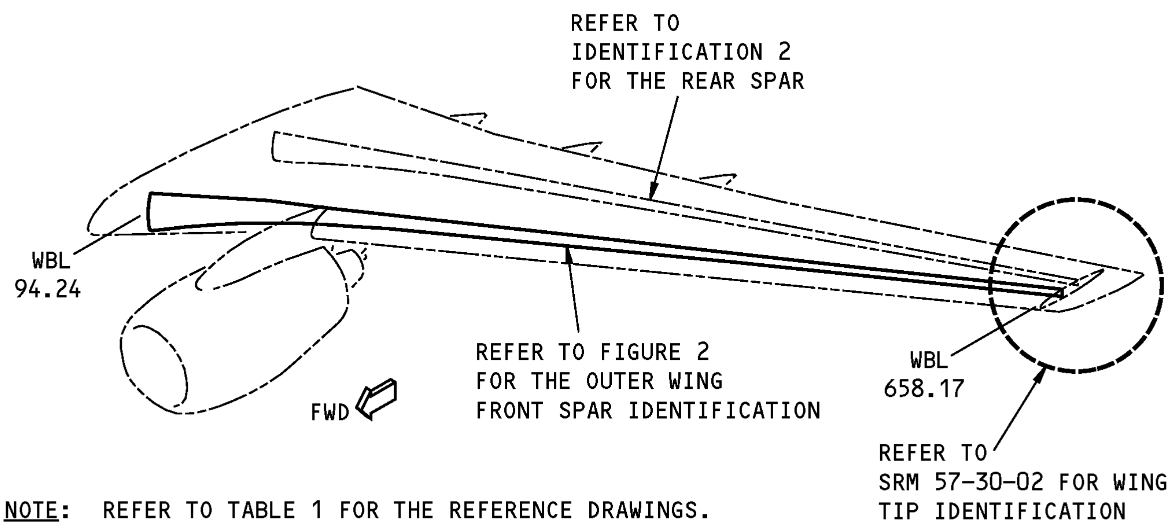
OUTER WING RIBS

A

**Outer Wing Rib Repairs**
**Figure 201**

## IDENTIFICATION 1 - OUTER WING FRONT SPAR STRUCTURE

### 1. Locator




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outer Wing Front Spar Location**
**Figure 1**

### 2. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 001A1001 | Wing Left - Product Collector |
| 111A2000 | Front Spar Installation - Wing Center Section |
| 112A0013 | Collector - Functional, Product, Front Spar, Section 12, Left |
| 112A2001 | Spar Installation - Front, Section 12, Wing |
| 112A2003 | Spar Assembly - Front, Section 12, Wing |
| 112A2004 | Leading Edge Support Installation - Front, Section 12, Wing |

### 3. Applicability

A. Identification 1 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 1 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

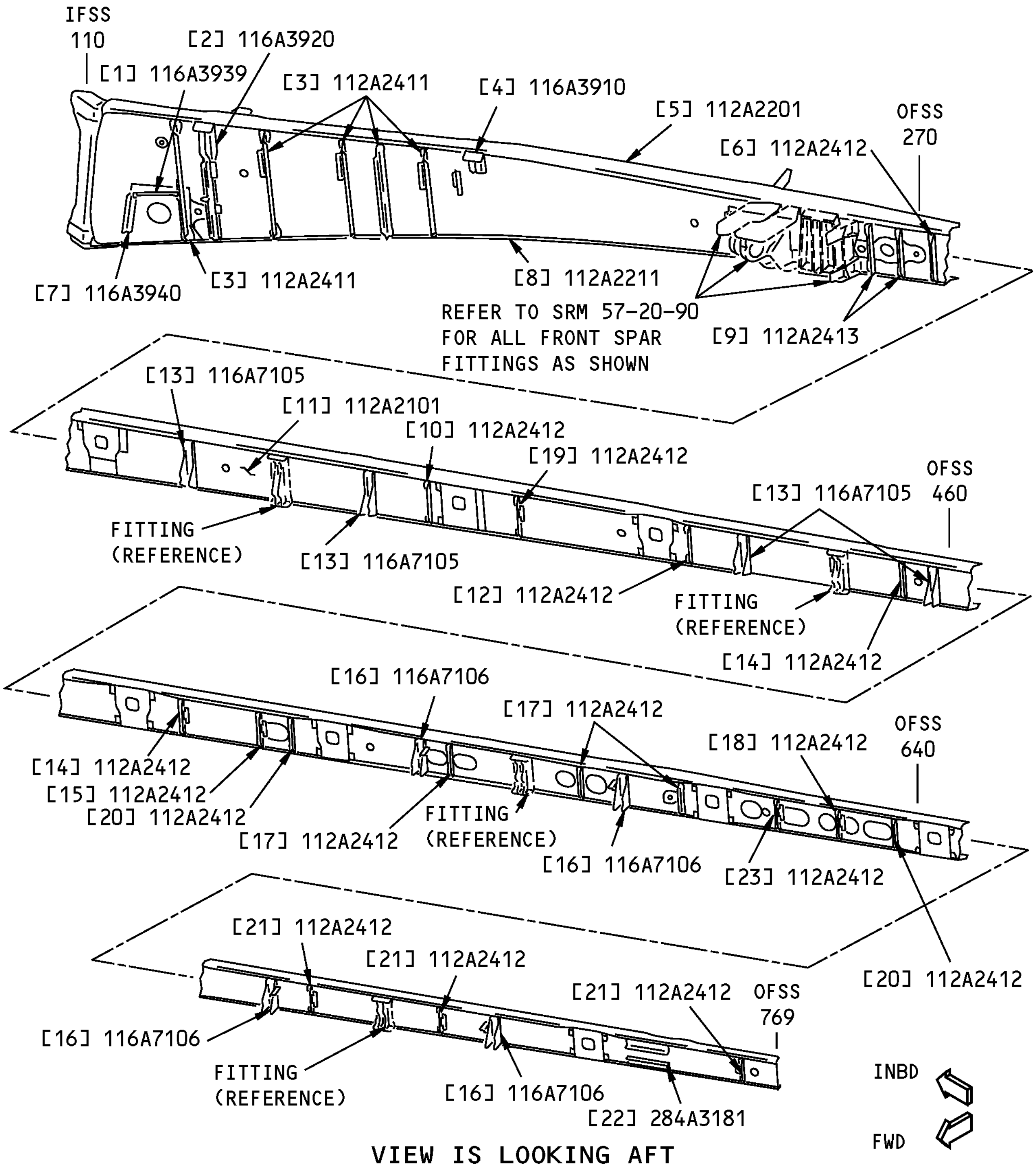
B. List of Materials:

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Angle Support | | 7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [2] | Actuator Support | | 7050-T7451 forged block as given in BMS 7-186. Refer to production drawing for the machined areas (Grain direction controlled part) | |
| [3] | Stiffener (10) | | 7050-T7451 plate as given in BMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [4] | Actuator Support | | 7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [5] | Chord, Upper | | BAC1514-3096 7150-T77511 extrusion as given in BMS 7-306<br>BAC1514-3105 7150-T77511 extrusion as given in BMS 7-306 | Cum Line 001 to 777<br>CUM LINE 778 and on |
| [6] | Stiffener | | BAC1503-100017 7075-T6511 extrusion as given in QQ-A-250/11 | |
| [7] | Angle Support | 0.063 (1.60) | 2024-T42 sheet as given in QQ-A-250/5 | |
| [8] | Chord, Lower | | BAC1514-3095 2224-T3511 extrusion as given in BMS 7-255 | |
| [9] | Stiffener (5) | | 7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [10] | Stiffener | | BAC1503-100130 7075-T6511 extrusion as given in QQ-A-200/11 | |
| [11] | Web | | 2324-T39 plate as given in BMS 7-254. Refer to the production drawing for the machined areas | |
| [12] | Stiffener | | BAC1503-100683 7075-T6511 extrusion as given in QQ-A-250/11 | |
| [13] | Support Bracket (4) | | BAC1509-100662 7075-T3511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined areas. (Optional: 7050-T7451 as given given in AMS 4050) | |
| [14] | Stiffener (2) | | BAC1514-1655 7075-T6511 extrusion as given in QQ-A-250/11 | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [15] | Stiffener | | BAV1503-100624 7075-T6511 extrusion as given in QQ-A-200/11 | |
| [16] | Bracket Assembly (4) | | | |
| | Bracket | | BAC1509-100662 7075-T73511 extrusion as given in QQ-A-200/11 (Optional: 7050-T7451 as given given in AMS 4050) | |
| | Bracket | 0.050 (1.27) | 6013-T6 sheet as given in AMS 4347 | |
| [17] | Stiffener (3) | | BAC1503-100607 7075-T6511 extrusion as given in QQ-A-250/11 | |
| [18] | Stiffener | | BAC1503-100671 7075-T6511 extrusion as given in QQ-A-250/11 | |
| [19] | Stiffener | | BAC1514-447 7075-T6511 extrusion as given in QQ-A-250/11 | Left side only |
| [20] | Stiffener(2) | | BAC1503-100387 7075-T6511 extrusion as given in QQ-A-200/11. | |
| [21] | Stiffener (3) | | BAC1503-100413 7075-T6511 extrusion as given in QQ-A-250/11 | |
| [22] | Angle (2) | | BAC1503-100796 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [23] | Stiffener | | BAC1503-100787 7075-T6511 extrusion as given in QQ-A-200/11 | |
| [24] | Stiffener | | 7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Optional: 7050-7451 as given in AMS 7–323, Type 1) (Grain direction controlled part). | |
| [25] | Rib Post (10) | | 7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [26] | Rib Post (2) | | 7050-T7451 plate as given in BMS 7-323, Type 1 (Optional: 7050-T7451 plate as given in AMS 4050) (Grain direction controlled part) | |
| [27] | Rib Post (9) | | 7050-T7451 plate as given in BMS 7-323, Type 1 (Grain direction controlled part) | |
| [28] | Rib Post (1) | | 7050-T7451 plate as given in BMS 7-323, Type 1 (Grain direction controlled part) | Left side only |
| [29] | Slat Can (7) | | D357-T6 investment casting as given in BMS 7-330. Refer to the production drawing for the machined areas | |

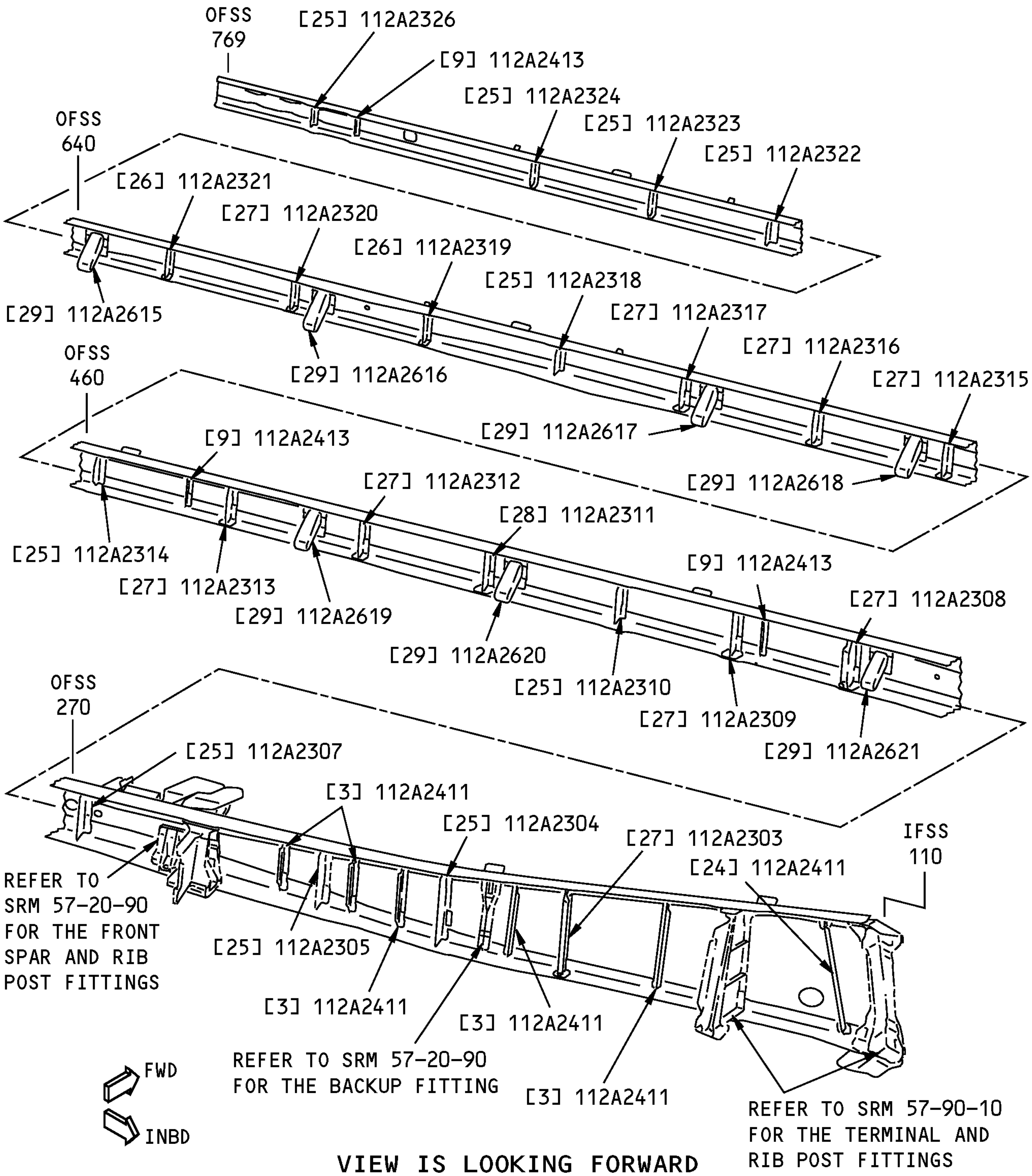
*[1] Note: T = Pre-manufactured thicknesses in inches (millimeters).




NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

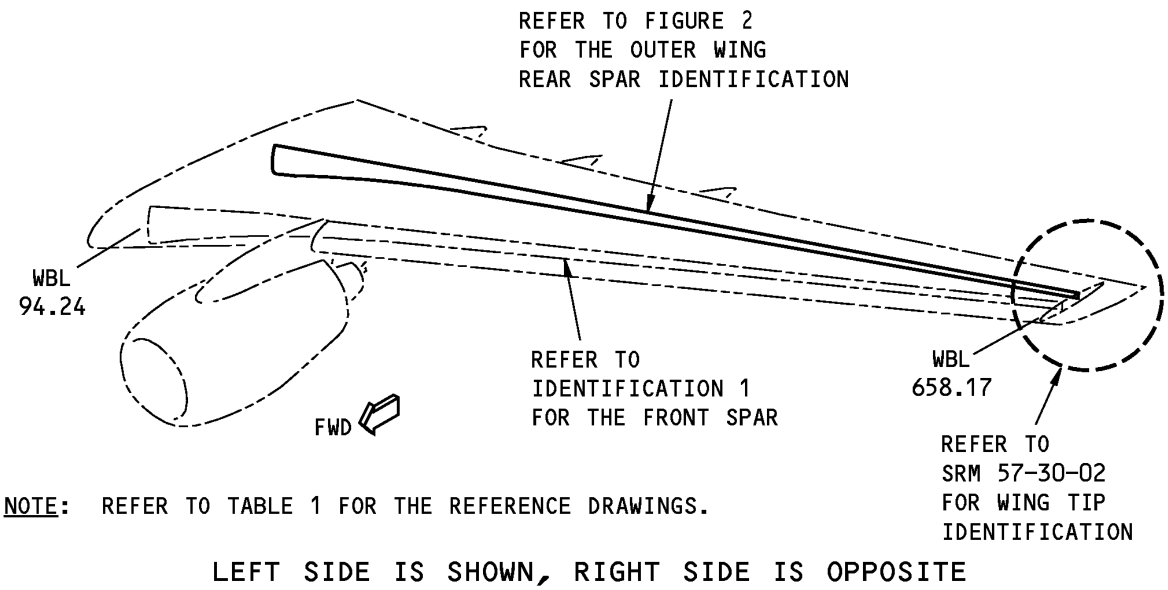
**Outer Wing Front Spar Identification**
**Figure 2 (Sheet 1 of 2)**




**Outer Wing Front Spar Identification**
**Figure 2 (Sheet 2 of 2)**

# IDENTIFICATION 2 - OUTER WING REAR SPAR

## 1. Locator




**Outer Wing Rear Spar Location**
**Figure 1**

## 2. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 001A1001 | Wing Left - Product Collector |
| 112A0011 | Collector - Functional, Product, Rear Spar, Section 12, Left |
| 112A1001 | Spar Installation - Rear Wing |
| 112A1003 | Spar Assembly - Rear Wing |

## 3. Applicability

A. Identification 2 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

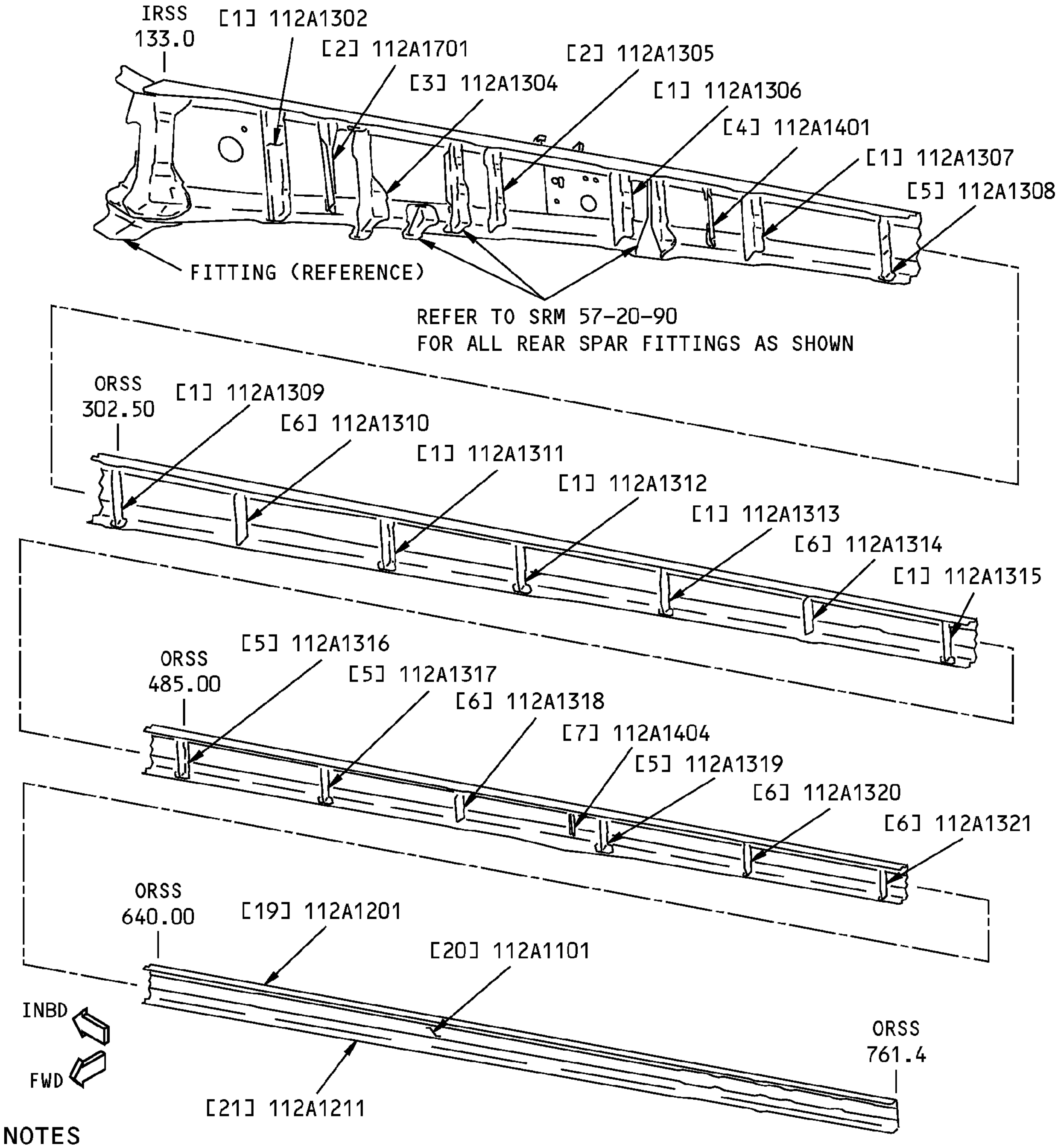
B. List of Materials

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Rib Post (8) | | 7050-T7451 machined plate as given in BMS 7-323, Type 1 Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [2] | Rib Post (2) | | 7050-T7451 machined plate as given in BMS 7-323, Type III Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [3] | Rib Post | | 7050-T7452 forged block as given in AMS 4050, Type III Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | YA001-YC003, YE001-YG999 |
| | | | 7050-T7452 forged block as given in BMS 7-214, Type II (Optional: 7050-T7452 forged block as given in AMS 4050, Type III) (Grain direction controlled part) | YC004-YD999 |
| [4] | Stiffener | | 7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | YA001-YC003 YE001-YG999 |
| | | | 7050-T7451 machined plate as given in BMS 7-323, Type I (Grain direction controlled part) | YC004-YD999 |
| [5] | Rib Post (4) | | 7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | YA001-YC003 YE001-YG999 |
| | | | 7050-T7451 machined plate as given in BMS 7-323, Type I (Grain direction controlled part) | YC004-YD999 |
| [6] | Rib Post (5) | | 7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [7] | Stiffener (9) | | 7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [8] | Stiffener (7) | | BAC1503-100917 7075-T6511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined thicknesses | |
| [9] | Stiffener (3) | | BAC1503-100066 7075-T6511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined thicknesses | |
| [10] | Bracket Assembly | | | |
| | Bracket | | 7075-T7351 machined plate, Class A as given in QQ-A-250/12. Refer to the production drawing for the machined thicknesses | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | Clip | 0.063 (1.6) | 6013-T6 formed sheet as given in AMS 4347 | |
| | Bracket | 0.063 (1.6) | 6013-T6 formed sheet as given in AMS 4347 | |
| | Bracket | 0.050 (1.27) | 6013-T6 formed sheet as given in AMS 4347 | |
| [11] | Stiffener (3) | | BAC1514-735 7075-T6511 extrusion as given in QQ-A-200/11 Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [12] | Stiffener (3) | | 7075-T73511 extruded bar as given in QQ-A-200/11 Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [13] | Bracket (3) | | 7075-T7351 machined plate as given in QQ-A-250/12. Refer to the production drawing for the machined thicknesses | |
| [14] | Stiffener (4) | | BAC1503-100161 7075-T6511 extrusion as given in QQ-A-200/11 Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [15] | Bracket | | 7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part) | |
| [16] | Bracket | 0.050 (1.27) | 6013-T6 formed sheet as given in AMS 4347 | |
| [17] | Stiffener (2) | | BAC1517-2769 7075-T6511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined thicknesses | |
| [18] | Bracket | 0.080 (2.03) | 6013-T6 formed sheet as given in AMS 4347 | |
| [19] | Chord, Upper | | BAC1514-3102 7150-T77511 formed extrusion as given in BMS 7-306 | |
| [20] | Web | | 2324-T39 machined plate as given in AMS 7-254. Refer to the production drawing for the machined thicknesses | |
| [21] | Chord, Lower | | BAC1514-3103 2224-T3511 formed extrusion as given in BMS 7-255 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



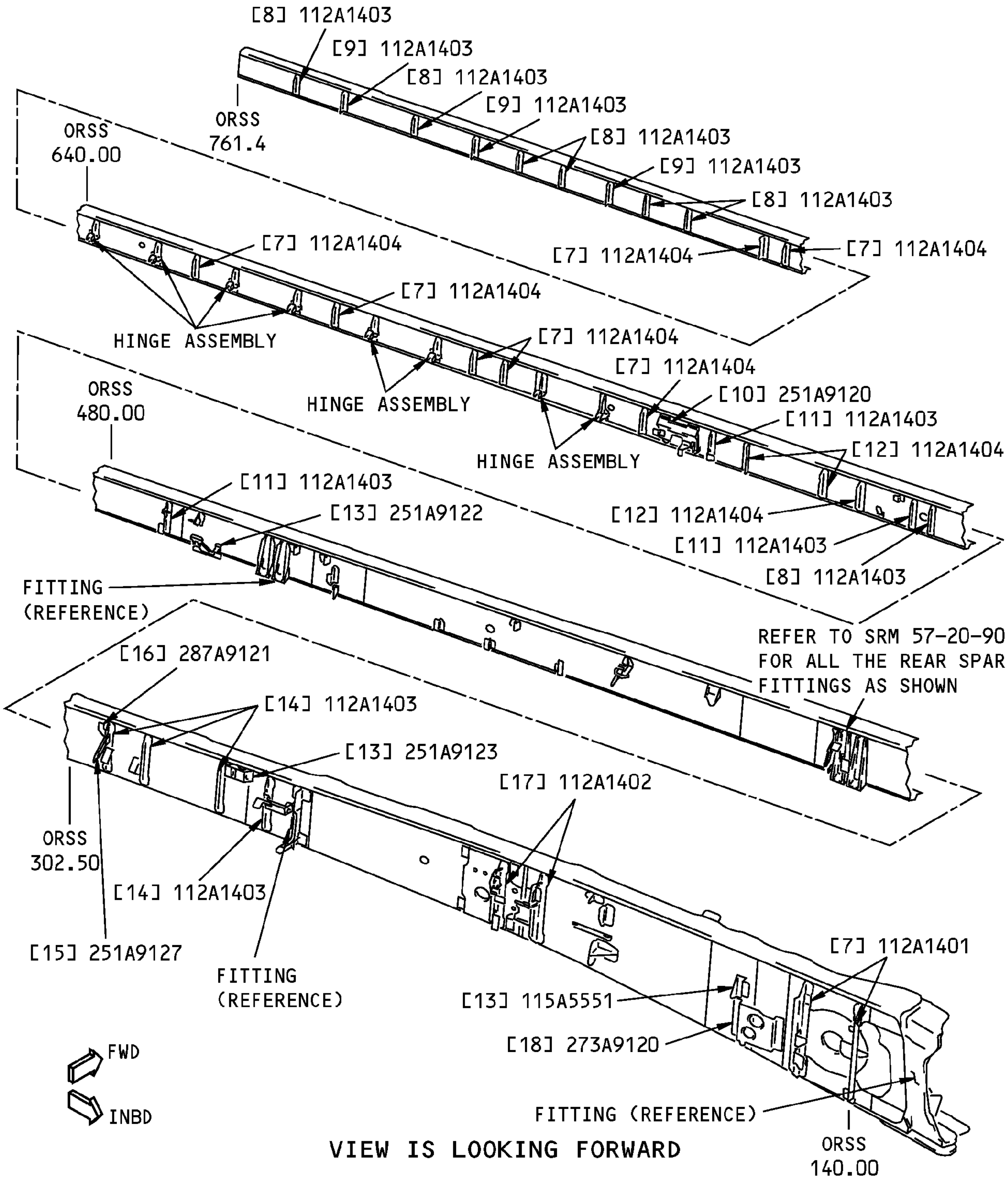

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

VIEW IS LOOKING AFT

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outer Wing Rear Spar Identification**
**Figure 2**




**Outer Wing Rear Spar Identification**
**Figure 3**

## ALLOWABLE DAMAGE 1 - OUTER WING FRONT SPAR

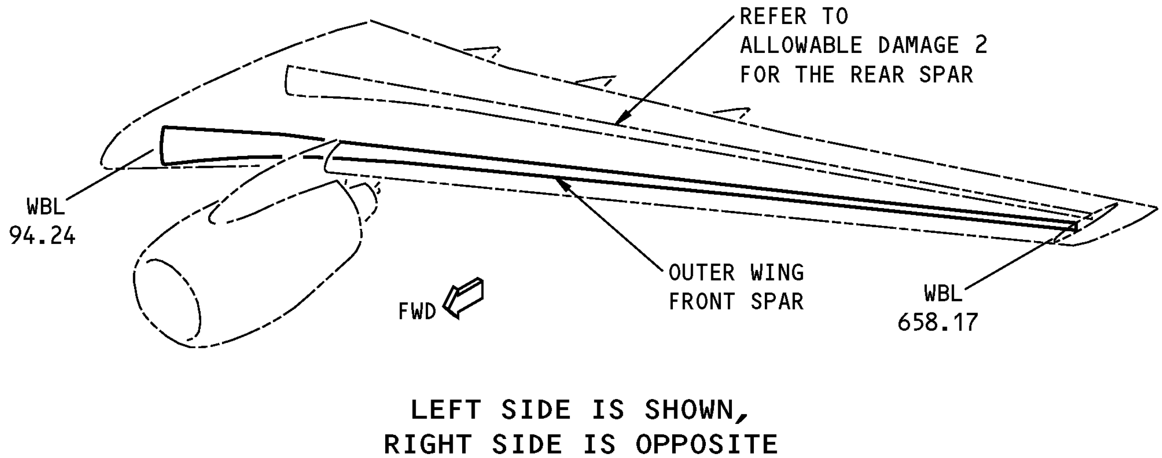
1. **Applicability**
   A. Allowable Damage 1 is applicable to damage on the outer wing front spar as shown in Location of the Outer Wing Front Spar, Figure 101/ALLOWABLE DAMAGE 1.
2. **General**
   A. Refer to Outer Wing Front Spar, Figure 102/ALLOWABLE DAMAGE 1 for the different parts of the front spar assembly.
   B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.
   C. Remove the damage.
      (1) Refer to 51-10-02 for the inspection and removal of damage.
      (2) Refer to 51-30-03 for possible sources of the abrasive and other materials you can use to remove the damage.
      (3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.
      (4) Make the surface texture roughness for all cut surfaces 125 microinches Ra or smoother.
   D. After you remove the damage, do the steps that follow:

   **WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

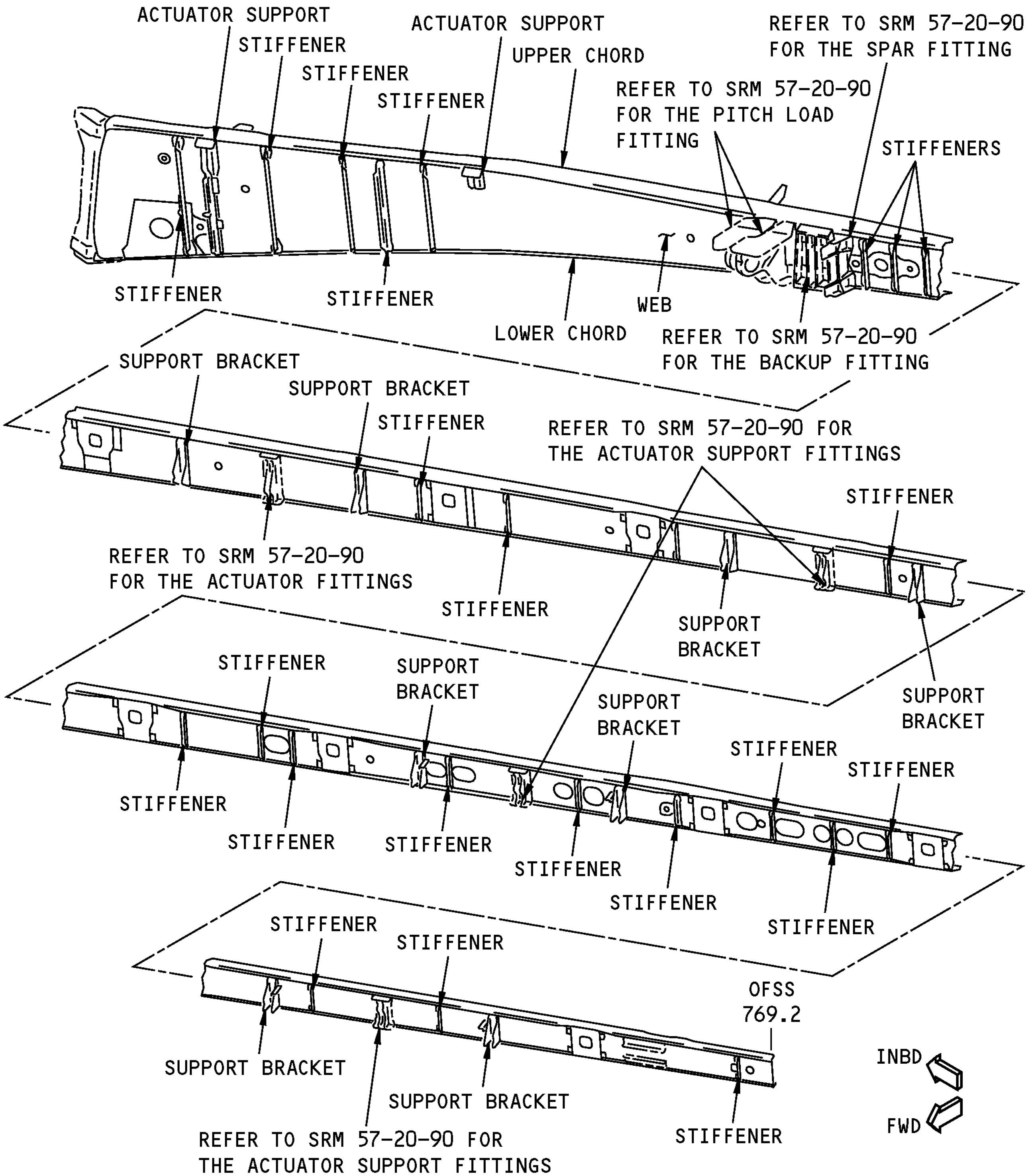
      (1) Flap peen or shot peen the reworked areas of the chords, web, and rib posts.
         (a) Refer to 51-20-06 for the shot peen intensity and shot number.
         (b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.
      (2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.
      (3) Apply one layer of BMS 10-20, Type II primer to the conversion coated, reworked areas. Refer to AMM 28-11-00/701.
      (4) Refer to 51-20-05 for sealing of fuel tanks.
      (5) Refer to 51-20-05 for the installation of fasteners in fuel tanks.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| **TYPE OF STRUCTURE** | **PARAGRAPH** |
| UPPER CHORD | 4.A |
| LOWER CHORD | 4.A |
| WEB | 4.B |
| STIFFENERS | 4.C |
| RIB POSTS | 4.C |
| SUPPORT BRACKETS | 4.D |
| ACTUATOR SUPPORTS | 4.D |

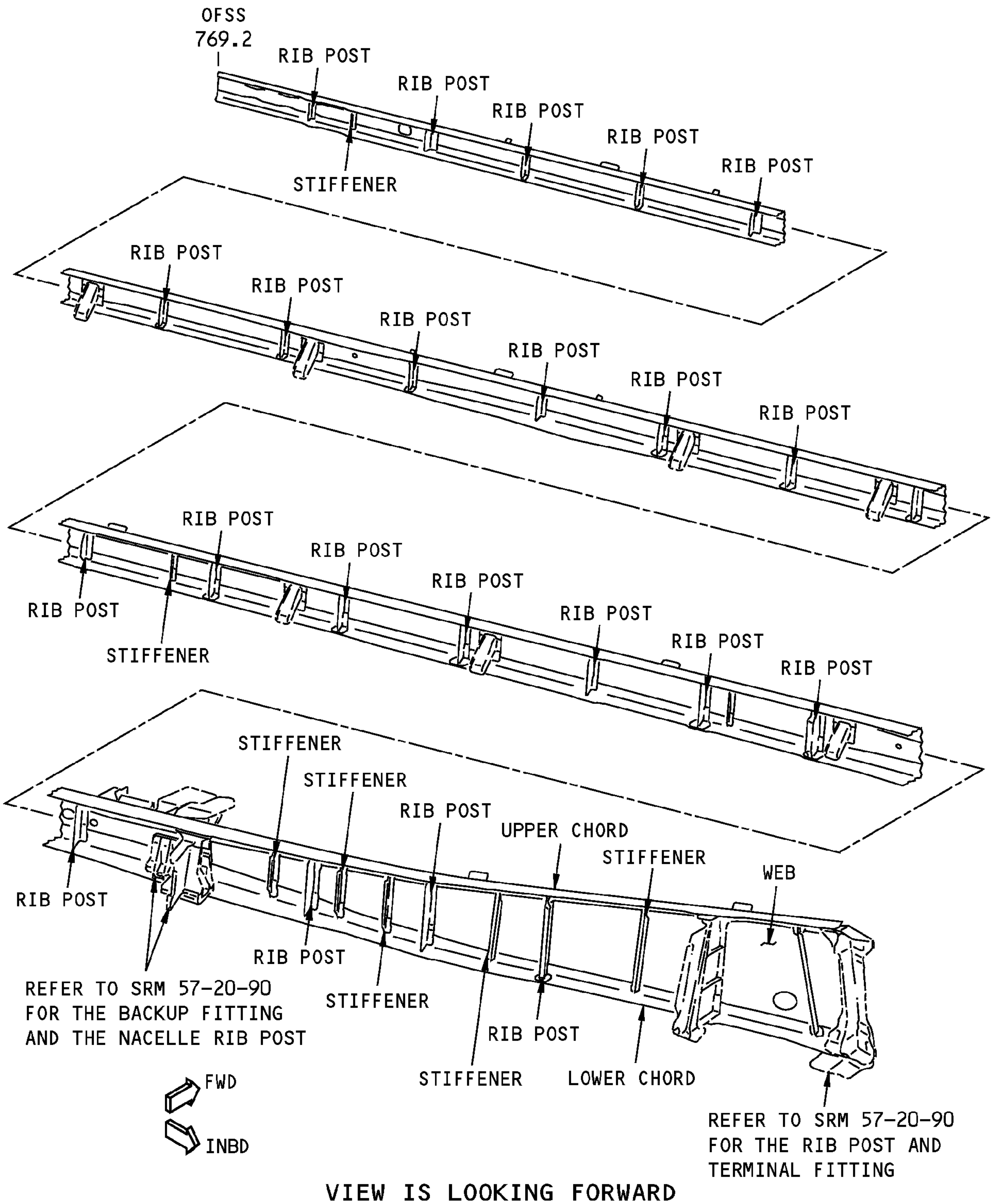



**Location of the Outer Wing Front Spar**
**Figure 101**




VIEW IS LOOKING AFT

**Outer Wing Front Spar**
**Figure 102 (Sheet 1 of 2)**




**Outer Wing Front Spar**
**Figure 102 (Sheet 2 of 2)**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Upper and Lower Chords

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E, and F.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage with one of the two procedures that follow:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , E, and H as applicable.

2) Drill a hole through the part at the damage location:

**NOTE**: Do not drill out the damage if the damage has been blended out before.

a) The hole must be:
- A maximum of 0.25 inch in diameter
- A minimum of 1.5 inches from a fastener hole, the edge of the part, or other damage.

b) Fill the hole with a 2117-T3 or 2117-T4 aluminum rivet installed wet with BMS 5-26 sealant.

(3) Dents are permitted if:

(a) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail G

(b) The edge of the damage is a minimum of 1.5 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) They are a maximum of 0.25 inch in diameter

(b) The edge of the damage is a minimum of 1.5 inches away from a fastener hole or other damage

(c) They are filled with a 2117-T3 or 2117-T4 aluminum rivet. Install the rivet with BMS 5-26 sealant.

C. Stiffeners and Rib Posts

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Support Brackets and Actuator Supports

(1) Cracks:

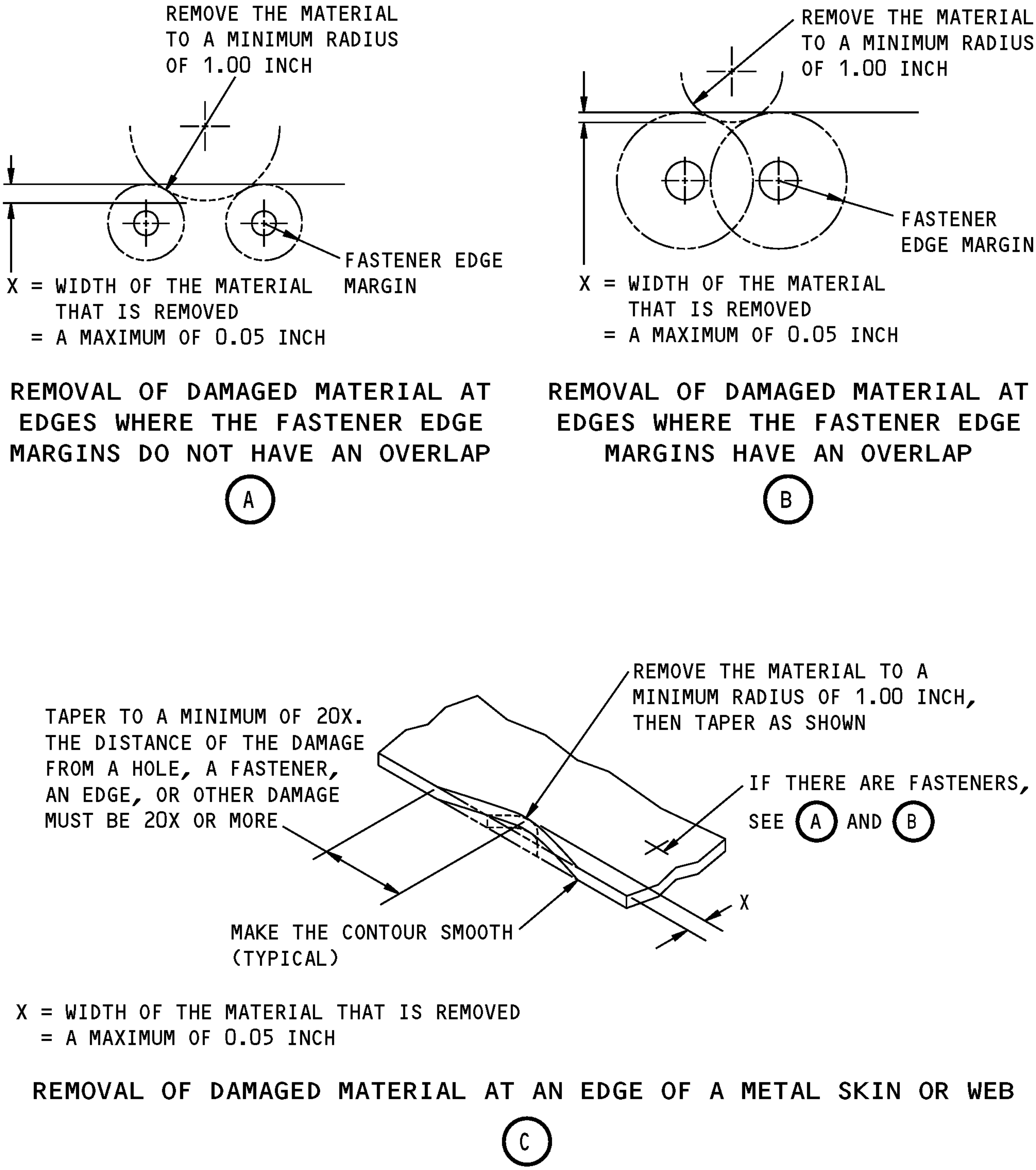
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

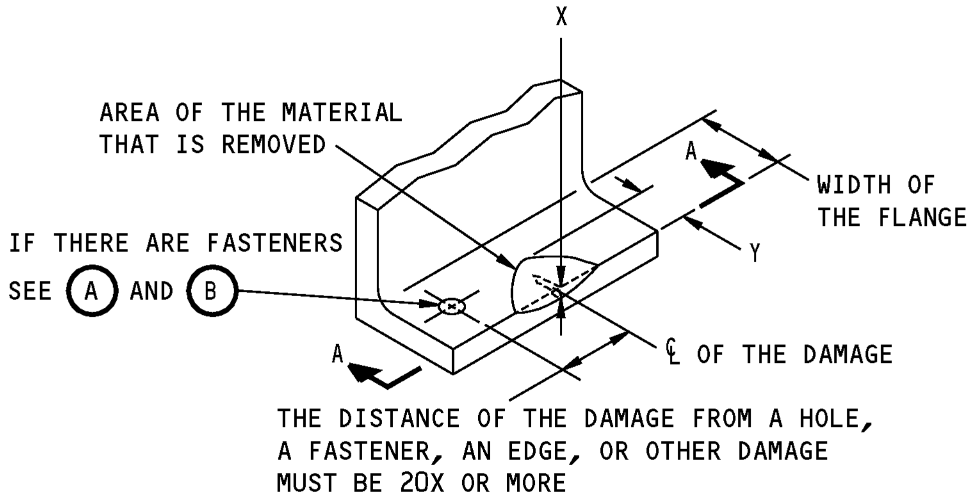
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.




**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 5)**

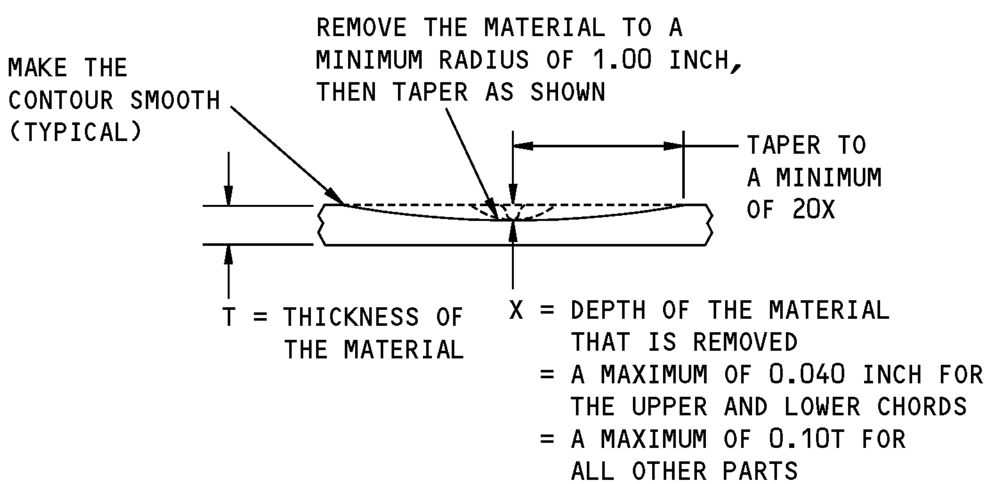



Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

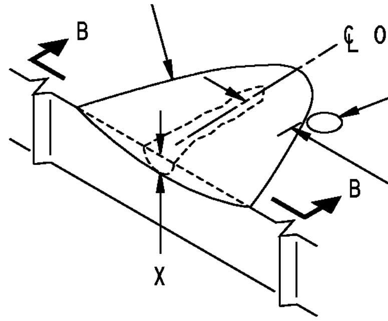
REMOVAL OF DAMAGED MATERIAL ON A SURFACE AT AN EDGE

D




A-A

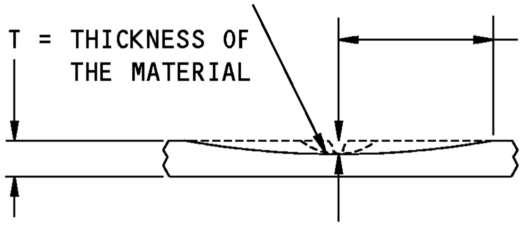
**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 5)**




NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

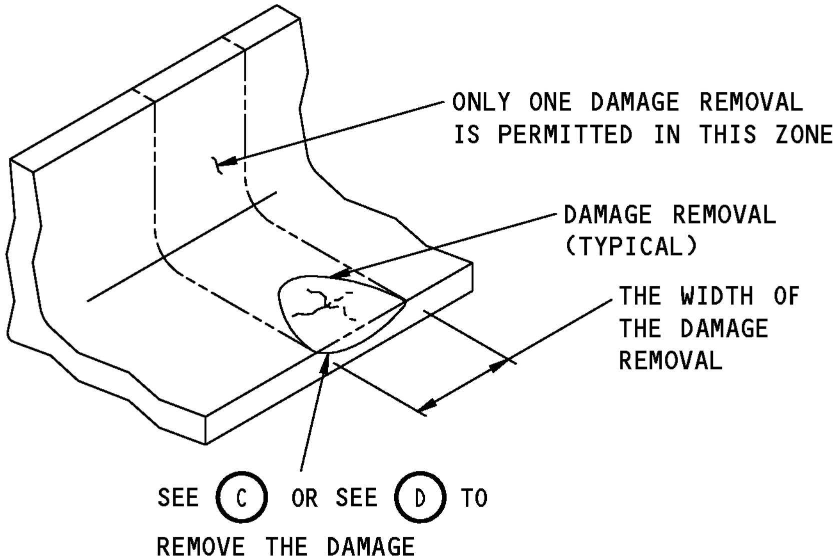
REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E



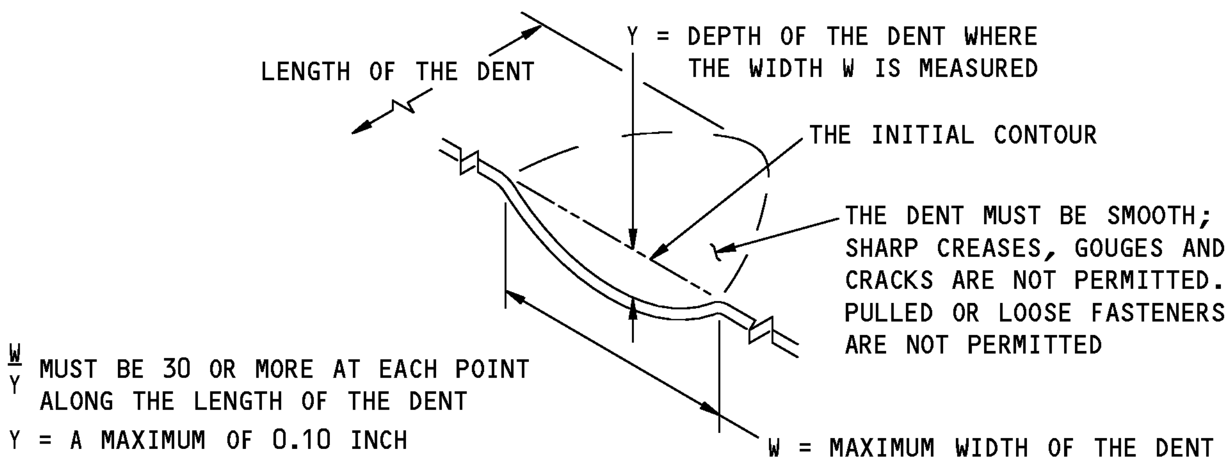

B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 5)**




DAMAGE LIMITS IN A CROSS SECTION OF A SPAR CHORD

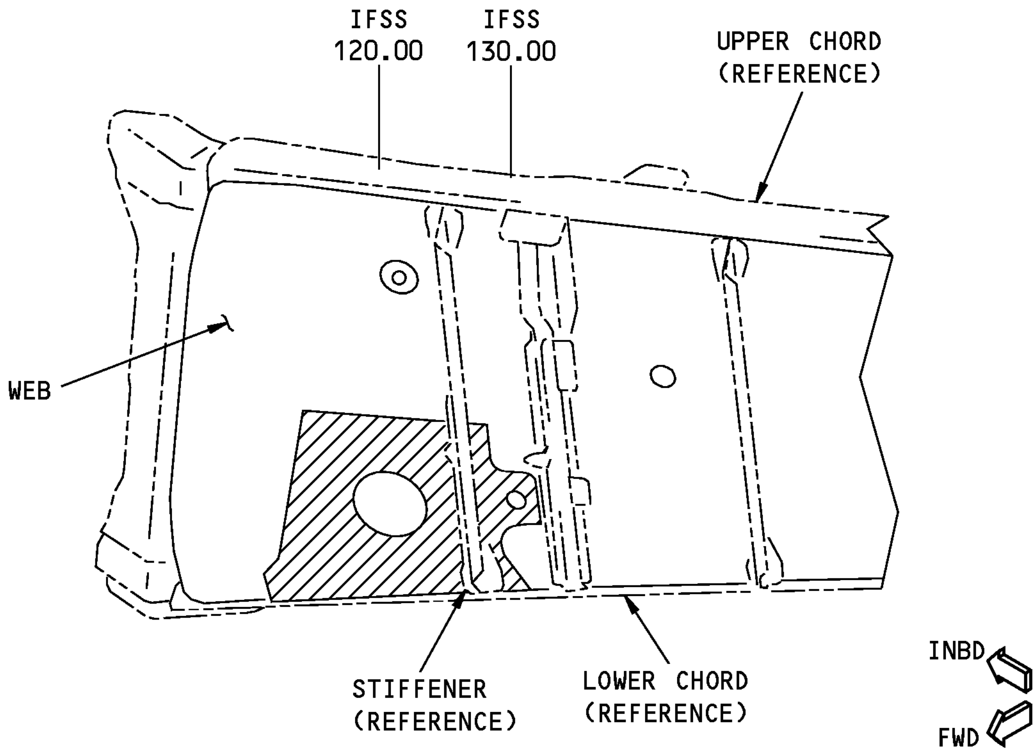
F




DENT THAT IS PERMITTED

G

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 5)**




ZONE SHOWS THE PAD-UP AREA OF THE WEB FOR THE TANK FUEL BOOST PUMP. THE CROSS-SECTIONAL AREA REMOVED FROM THIS ZONE MUST NOT BE MORE THAN 0.3 SQUARE INCHES. REFER TO FIGURE 103, DETAIL E FOR THE DEPTH LIMIT AND THE DAMAGE REMOVAL PROCEDURE.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H

**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 5)**

## ALLOWABLE DAMAGE 2 - OUTER WING REAR SPAR

1. **Applicability**

   A. Allowable Damage 2 is applicable to damage on the outer wing rear spar shown in Location of the Outer Wing Rear Spar, Figure 101/ALLOWABLE DAMAGE 2.

2. **General**

   A. Refer to Outer Wing Rear Spar, Figure 102/ALLOWABLE DAMAGE 2 for the different parts of the rear spar assembly.

   B. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

   C. Remove the damage.

      (1) Refer to 51-10-02 for the inspection and removal of damage.

      (2) Refer to 51-30-03 for possible sources of the abrasive and other materials you can use to remove the damage.

      (3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

   D. After you remove the damage, do the steps that follow:

   **WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

      (1) Flap peen or shot peen the reworked areas of the chords, web, and rib posts.

         (a) Refer to 51-20-06 for the shot peen intensity and shot number.

         (b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

      (2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

      (3) Apply one layer of BMS 10-20, Type II primer to the conversion coated, reworked areas. Refer to AMM 28-11-00/701.

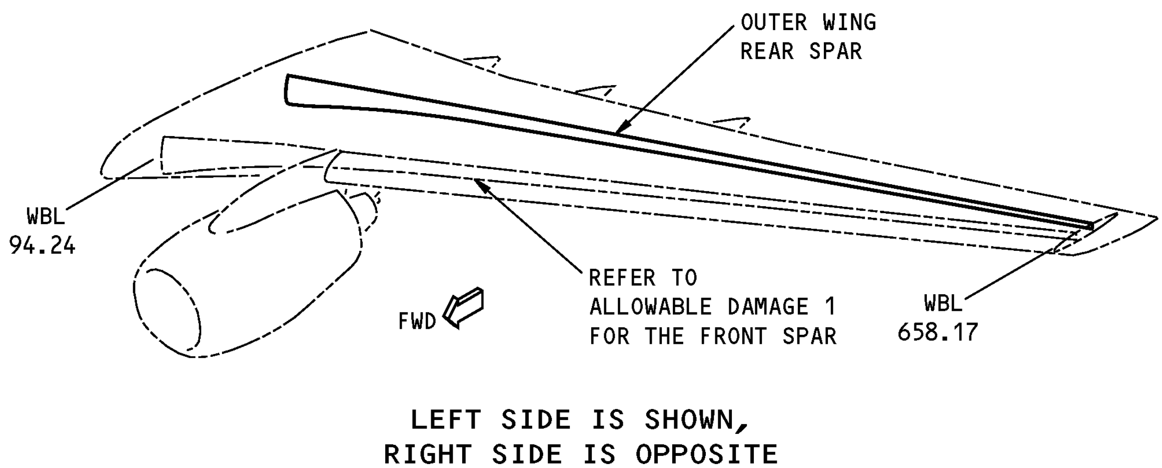
      (4) Refer to 51-20-05 for sealing of fuel tanks.

      (5) Refer to 51-20-05 for the installation of fasteners in fuel tanks.

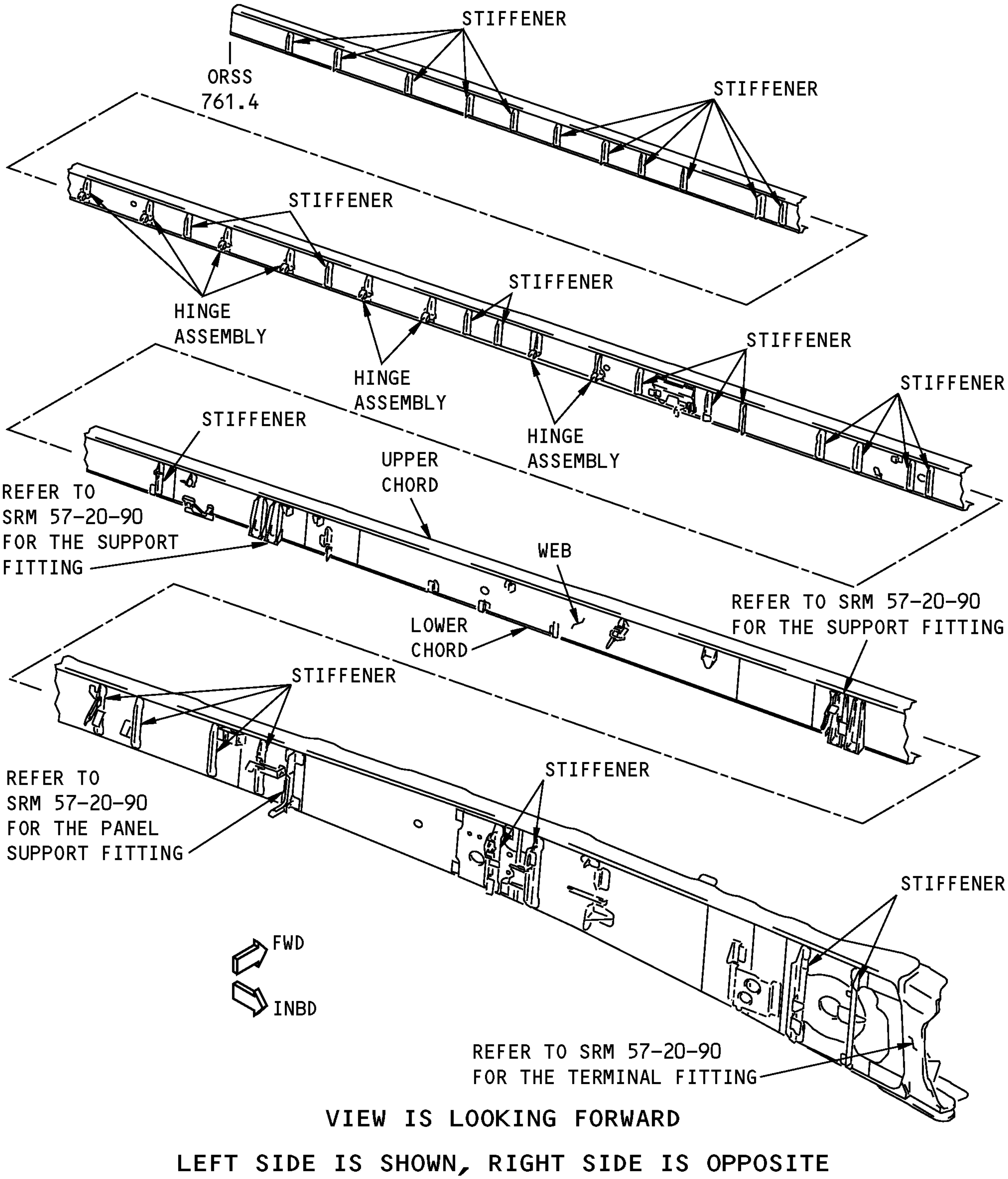
   E. Refer to Table 101/ALLOWABLE DAMAGE 2 for the references for the allowable damage limits of the different types of structure.

**Table 101**:

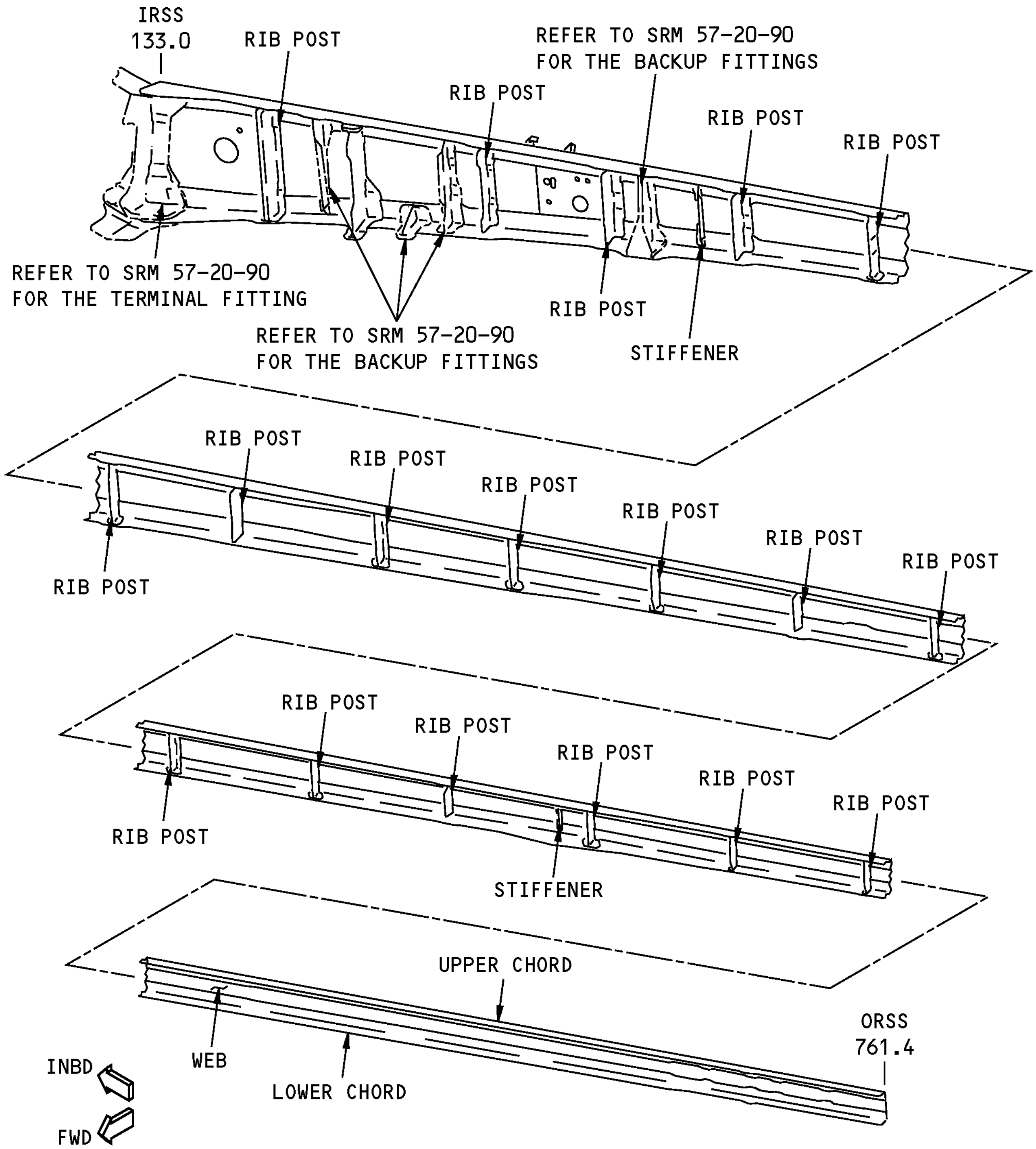
| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| **TYPE OF STRUCTURE** | **PARAGRAPH** |
| UPPER CHORD | 4.A |
| LOWER CHORD | 4.A |
| WEB | 4.B |
| STIFFENERS | 4.C |
| RIB POSTS | 4.C |
| HINGE ASSEMBLY | 4.D |




**Location of the Outer Wing Rear Spar**
**Figure 101**




**Outer Wing Rear Spar**
**Figure 102 (Sheet 1 of 2)**




VIEW IS LOOKING AFT

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outer Wing Rear Spar**
**Figure 102 (Sheet 2 of 2)**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| AMM 28-11-00/701 | Fuel Tanks - Cleaning and Painting |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Upper and Lower Chords

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , and F .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage with one of the two procedures that follows:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , E, and H as applicable.

2) Drill a hole through the part at the damage location:

**NOTE**: Do not drill out the damage if the damage has been blended out before.

a) The hole must be:

- A maximum of 0.25 inch in diameter
- A minimum of 1.5 inches from a fastener hole, the edge of the part, or other damage.

b) Fill the hole with a 2117-T3 or 2117-T4 aluminum rivet installed wet with BMS 5-26 sealant.

(3) Dents are permitted if:

(a) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail G

(b) The edge of the damage is a minimum of 1.5 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) They are a maximum of 0.25 inch in diameter

(b) The edge of the damage is a minimum of 1.5 inches away from a fastener hole or other damage

(c) They are filled with a 2117-T3 or 2117-T4 aluminum rivet. Install the rivet wet with 5-26 sealant.

C. Stiffeners and Rib Posts

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Hinge Assembly

(1) Cracks:

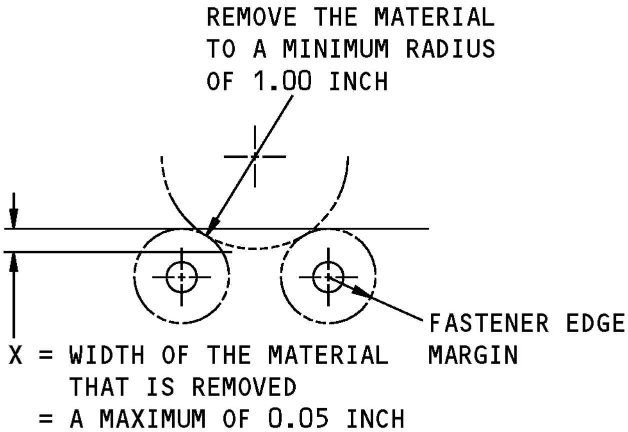
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

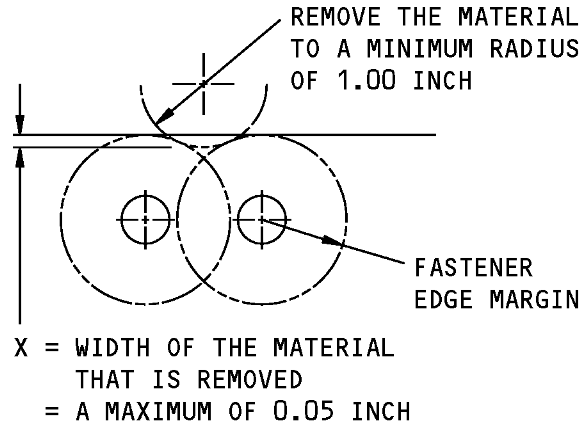
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



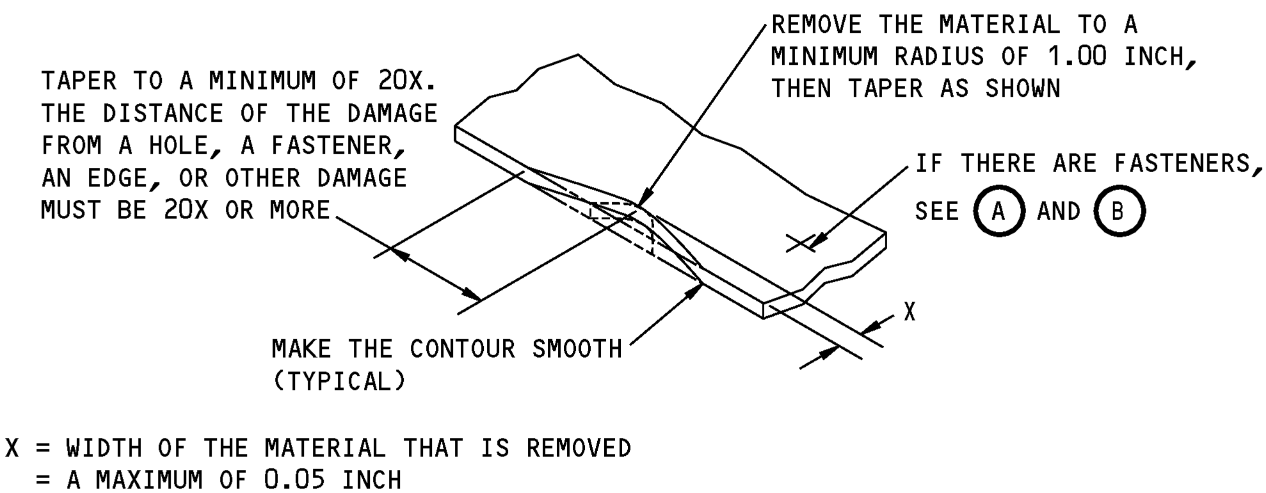

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

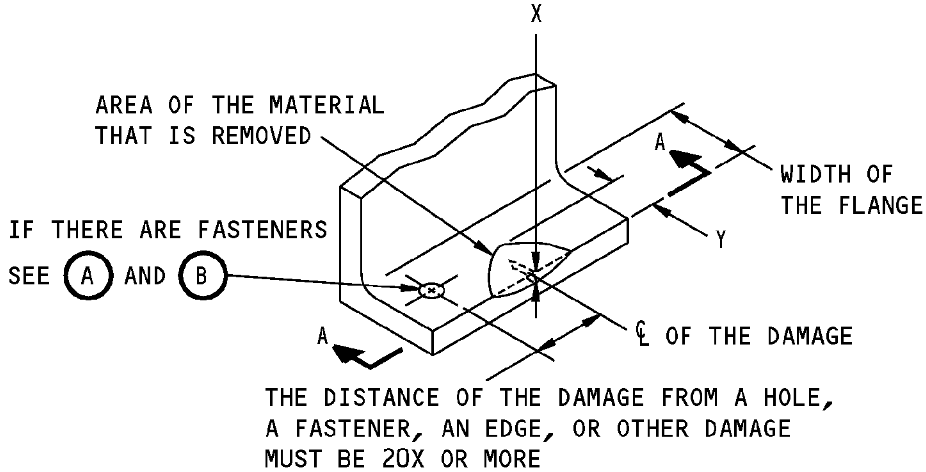
B




REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

C

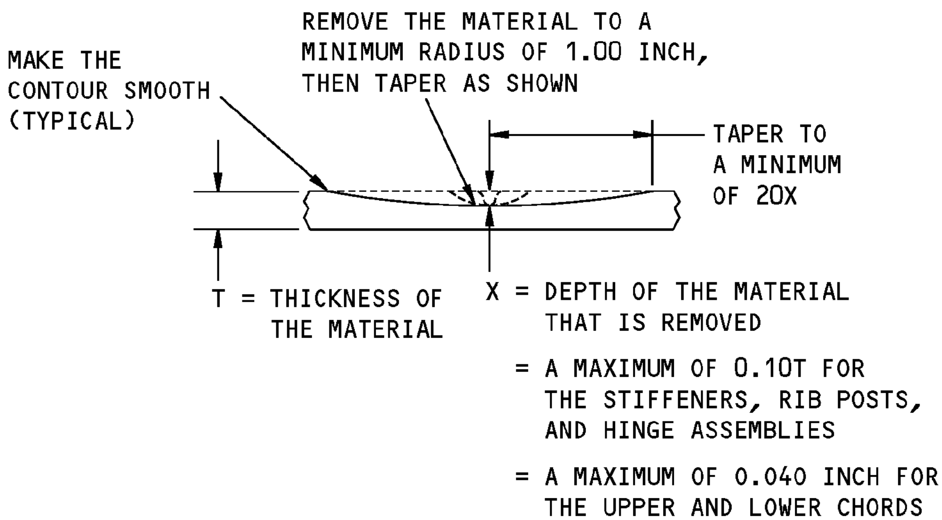
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 5)**




Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

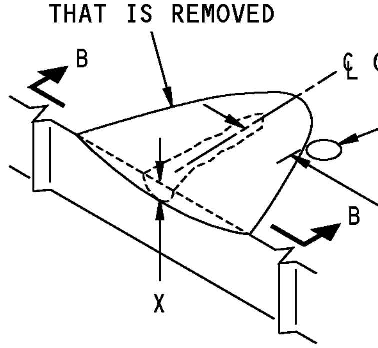
NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE
D




A-A

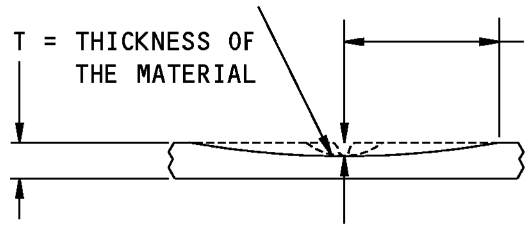
**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 5)**




NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

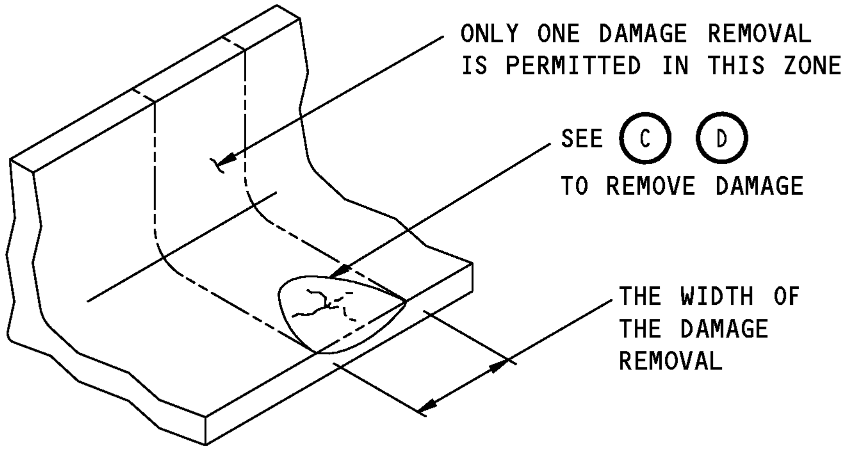
REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E



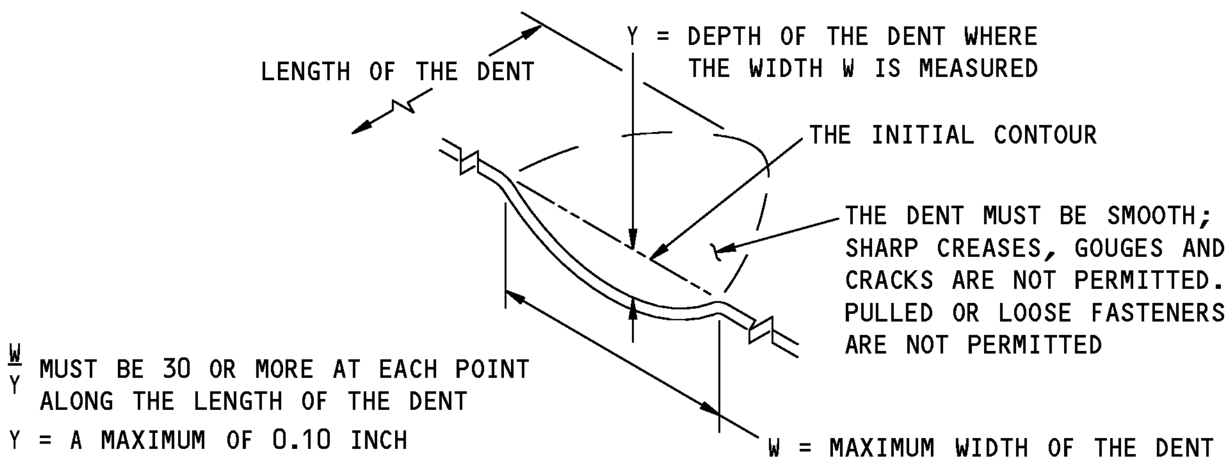

B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 5)**




DAMAGE LIMITS IN A CROSS SECTION OF A SPAR CHORD

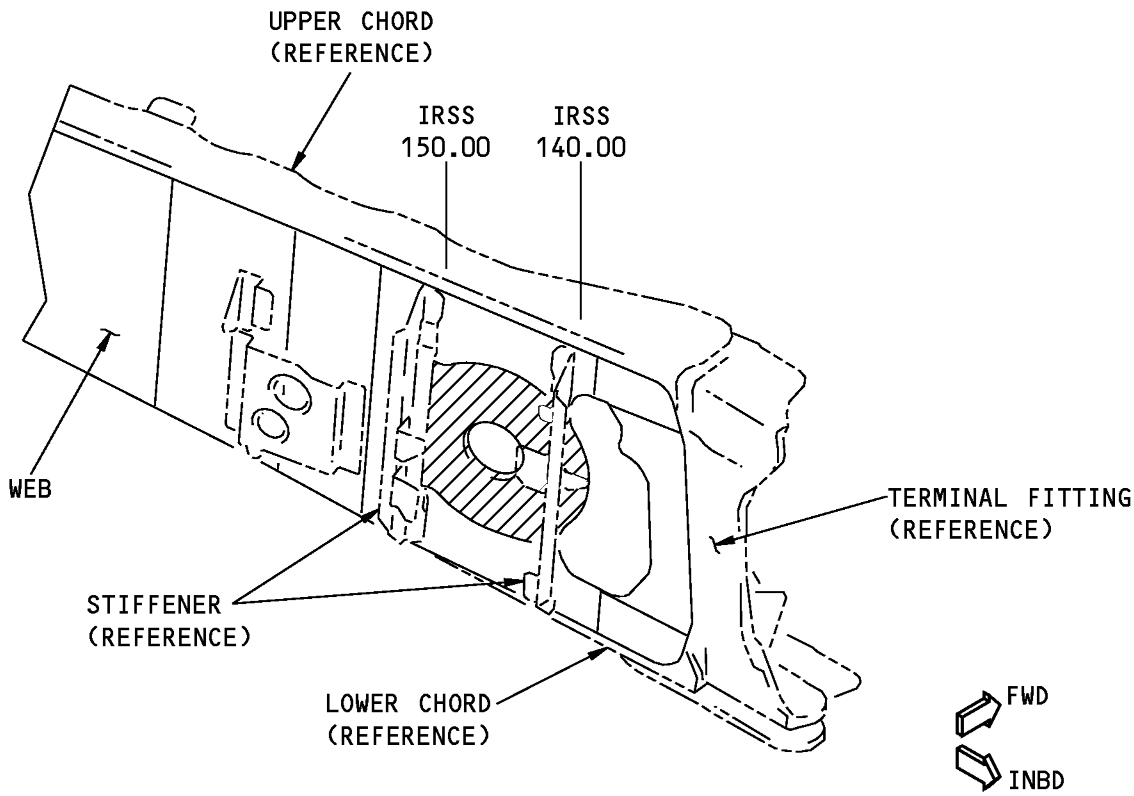
F




DENT THAT IS PERMITTED

G

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 5)**



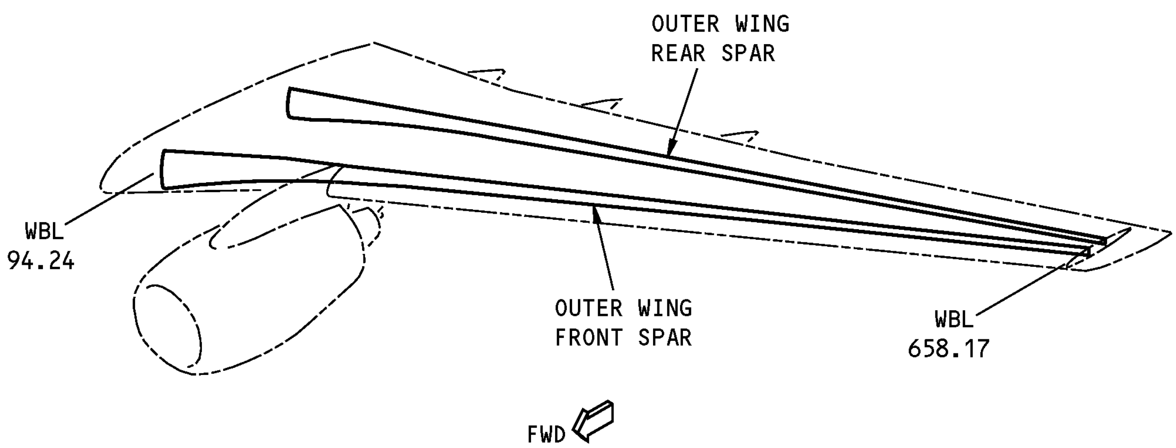

ZONE SHOWS THE PAD-UP AREA OF THE WEB FOR THE TANK FUEL BOOST PUMP. THE CROSS-SECTIONAL AREA REMOVED FROM THIS ZONE MUST NOT BE MORE THAN 1.0 SQUARE INCH. REFER TO FIGURE 103, DETAIL E FOR THE DEPTH LIMIT AND THE DAMAGE REMOVAL PROCEDURE.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H

**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 5)**

## REPAIR 1 - OUTER WING SPARS

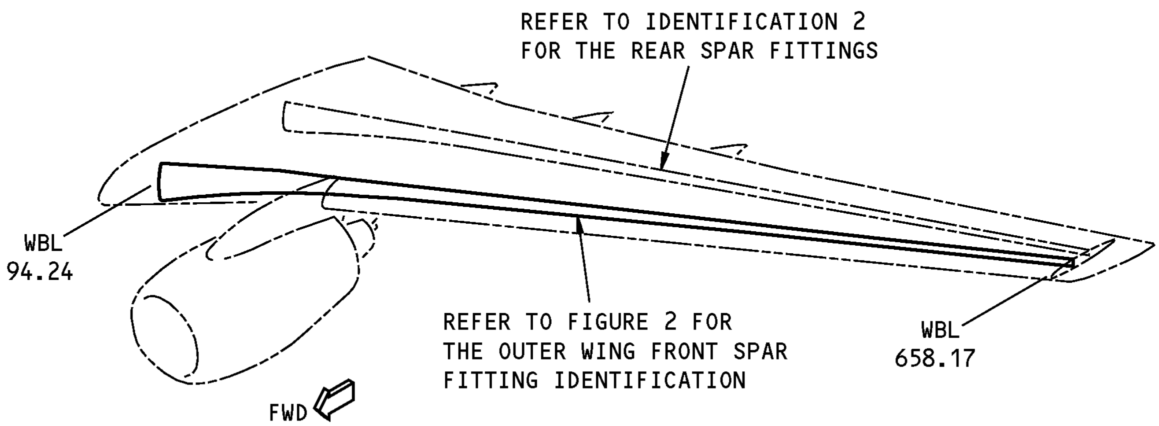



NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outer Wing Spar Repair**
**Figure 201**

## IDENTIFICATION 1 - OUTER WING FRONT SPAR FITTINGS



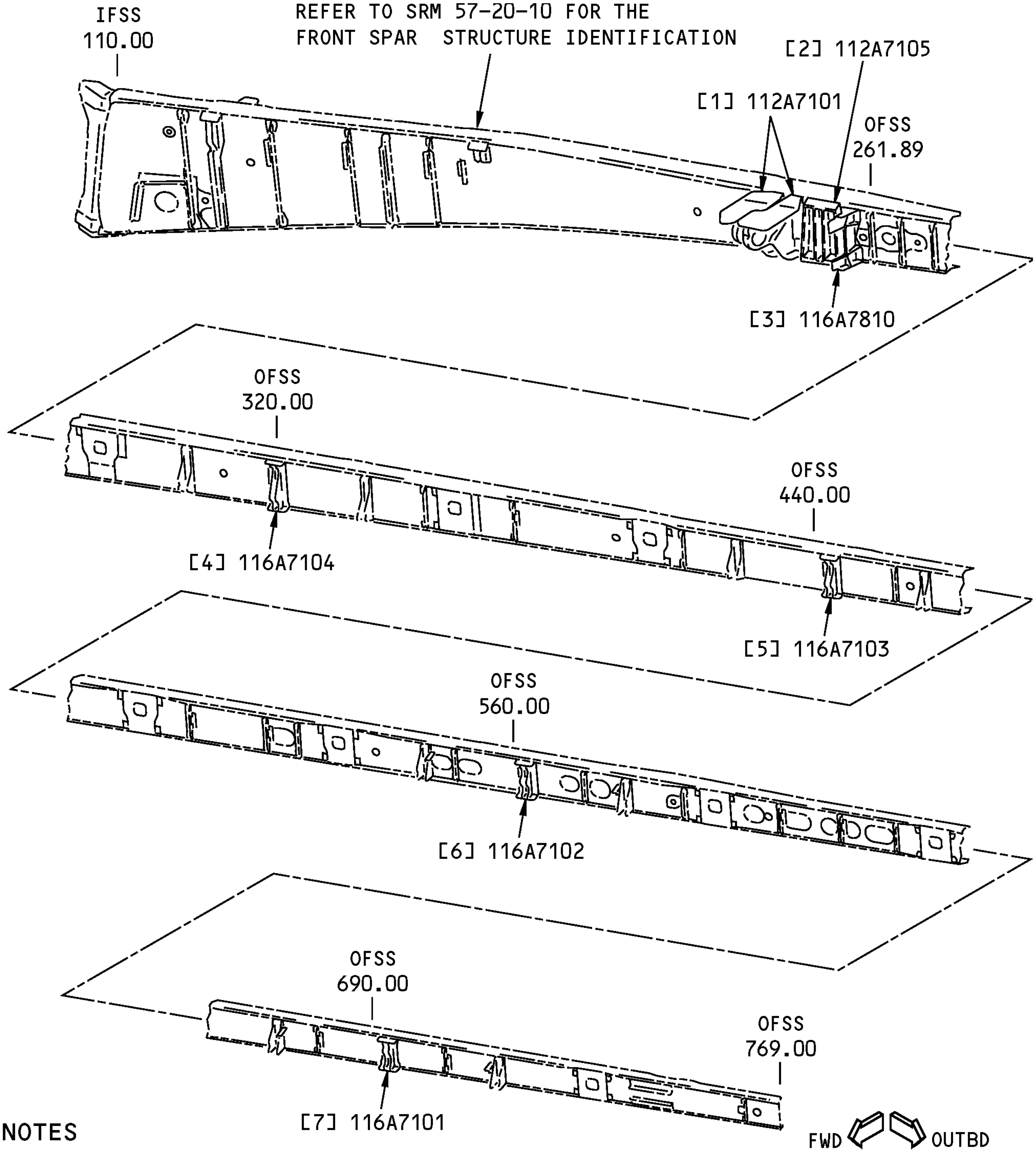

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outer Wing Front Spar Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A1001 | Wing Left - Product Collector |
| 001A1002 | Wing Right - Product Collector |
| 112A0013 | Collector - Functional, Product, Front Spar, Section 12, Left |
| 112A0014 | Collector - Functional, Product, Front Spar, Section 12, Right |
| 112A2003 | Front Spar Installation - Wing |
| 112A2501 | Terminal Fitting Installation - Front Spar, Wing BBL 70.85 |
| 112A2511 | Terminal Fitting - Front Spar, Wing to Body Joint |
| 112A7100 | Nacelle Support Fitting Installation - Front Spar |

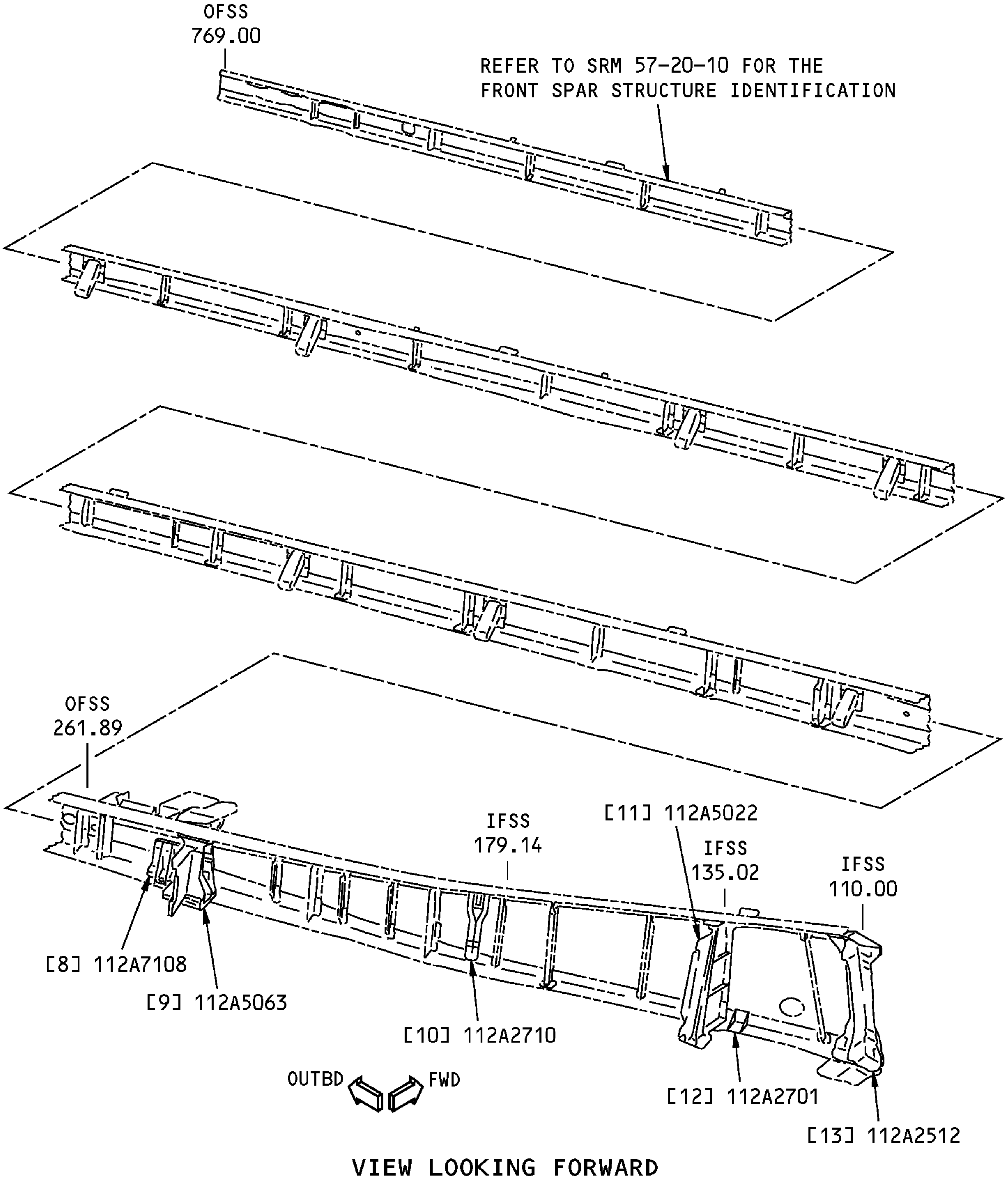



NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

VIEW LOOKING AFT

**Outer Wing Front Spar Fitting Identification**
**Figure 2 (Sheet 1 of 2)**



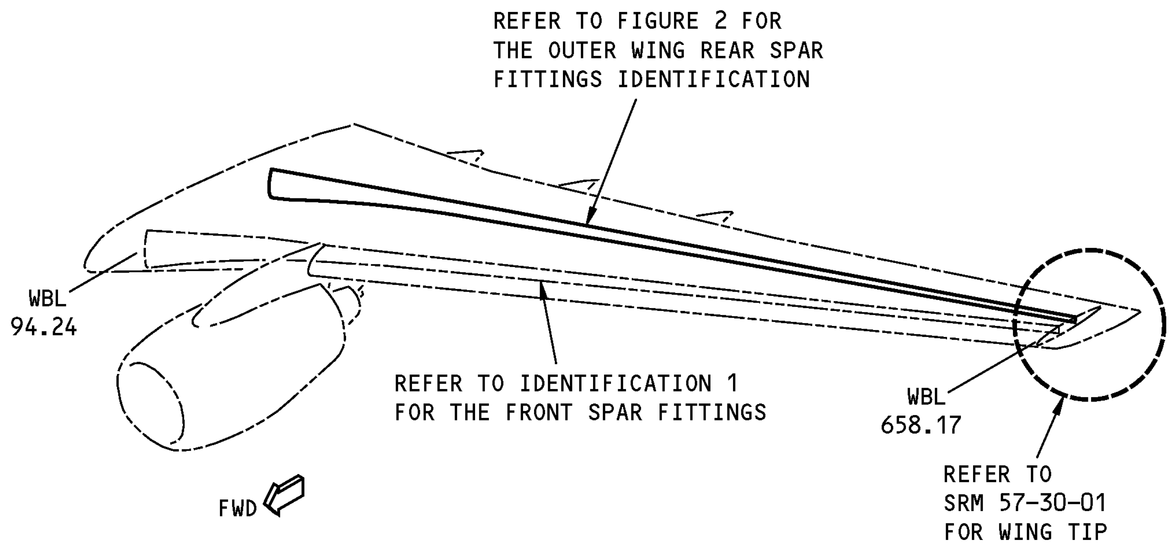

**Outer Wing Front Spar Fitting Identification**
**Figure 2 (Sheet 2 of 2)**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Nacelle Support Fitting, Outboard and Inboard Pitch Load (2) | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas. | |
| [2] | Nacelle Support Backup Fitting, Outboard Side Load | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas | |
| [3] | Spar Fitting, Fixed Leading Edge, Canted Rib | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas | |
| [4] | Actuator Support Fitting, Fixed Leading Edge at FSS 85.19 | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas | |
| [5] | Actuator Support Fitting at FSS 202.45 | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas | |
| [6] | Actuator Support Fitting, Fixed Leading Edge at FSS 322.71 | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas | |
| [7] | Actuator Support Fitting at FSS 451.97 | | 7050-T7451 plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas | |
| [8] | Nacelle Support Backup Fitting, Side Link | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas | |
| [9] | Nacelle Rib Post | | 7050-T73 forging as given in BMS 7-186. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [10] | Backup Fitting at IFSS 185 | | 7050-T7451 machined plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [11] | Jack Fitting | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [12] | Backup Fitting at IFSS 132 | | 7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [13] | Terminal Fitting, Front Spar | | 7050-T74 forging as given in BMS 7-214. Refer to the production drawing for the machined areas (Grain direction controlled part) | Cum Line Numbers 001 to 744 |
| | | | 7050-T7452 forging as given in BMS 7-214. Refer to the production drawing for the machined areas (Grain direction controlled part) | Cum Line Number 745 and on |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - OUTER WING REAR SPAR FITTINGS



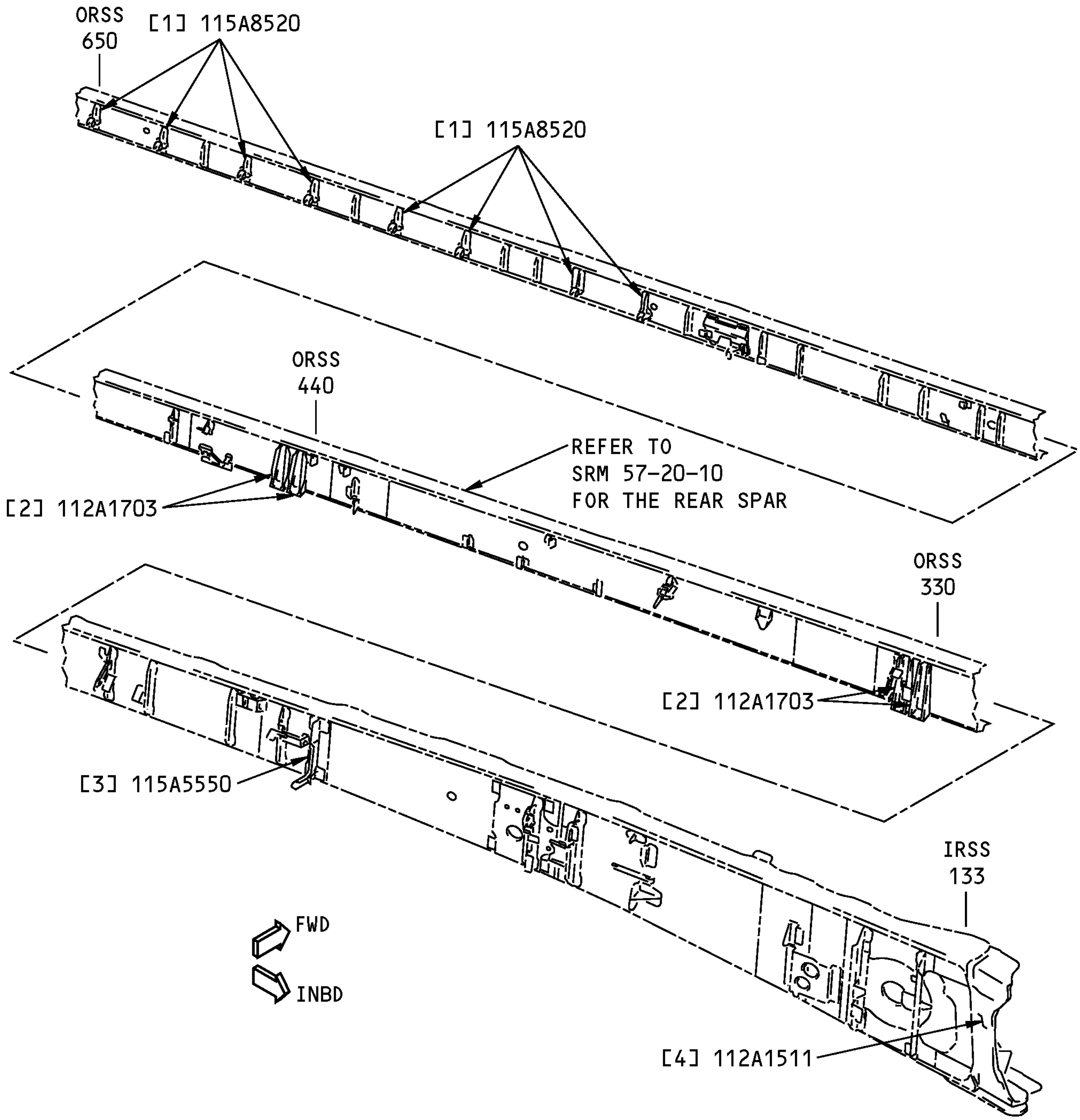

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outer Wing Rear Spar Fittings Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 001A1001 | Wing Left - Product Collector |
| 001A1002 | Wing Right - Product Collector |
| 112A0011 | Collector - Functional, Product, Rear Spar, Section 12, left |
| 112A0012 | Collector - Functional, Product, Rear Spar, Section 12, Right |
| 112A1003 | Spar Assembly - Rear, Wing |
| 112A1501 | Terminal Fitting Installation - Rear Spar, Wing BBL 70.85 |



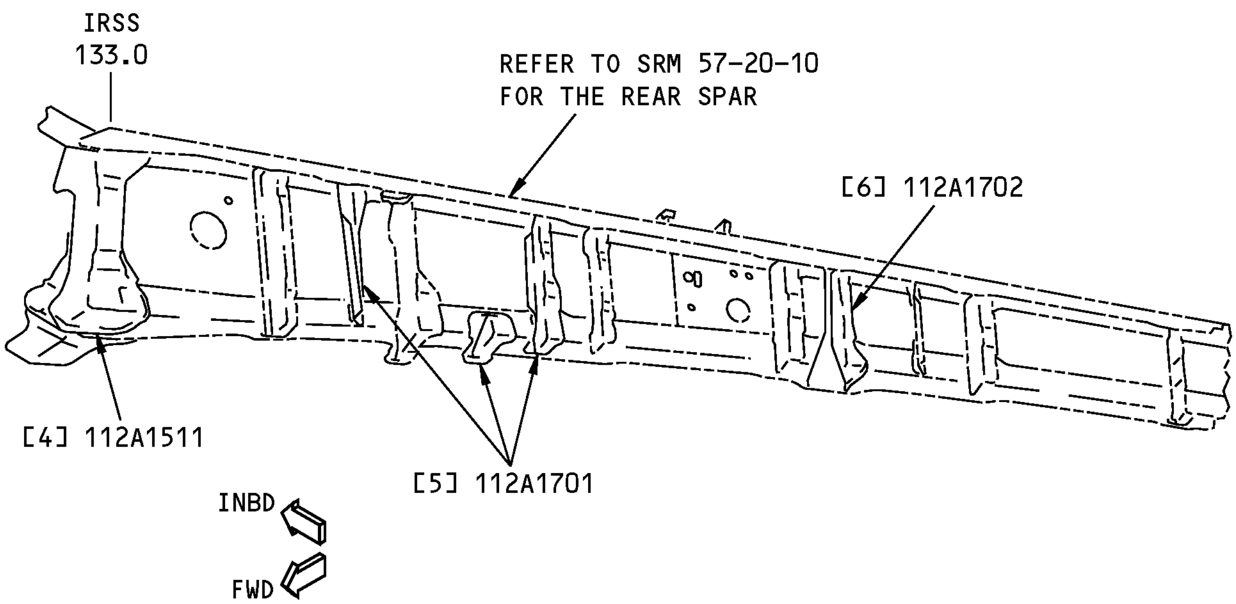

VIEW IS LOOKING FORWARD

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Outer Wing Rear Spar Fittings Identification**
**Figure 2 (Sheet 1 of 2)**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outer Wing Rear Spar Fittings Identification**
**Figure 2 (Sheet 2 of 2)**

**Table 2**:

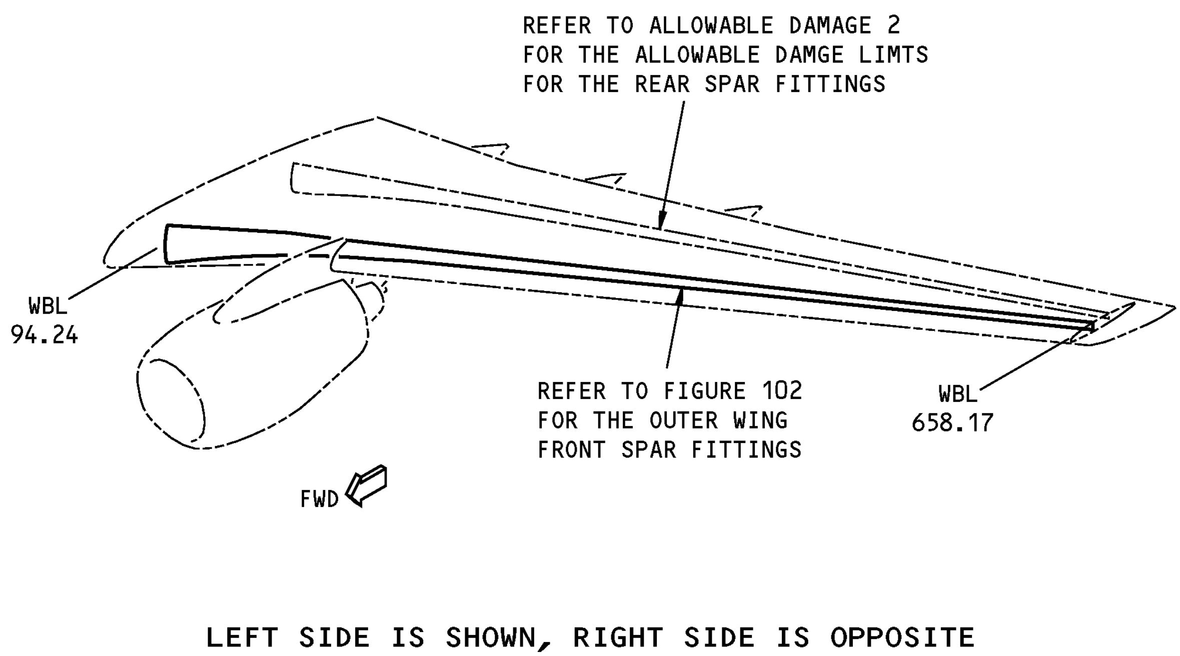
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Support Fitting, Aileron Balance Panel | | 7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [2] | Support Fitting, Flap | | 7050-T7451 machined plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined areas (Grain direction controlled part) | |
| [3] | Support Bracket, Inboard Fixed Trailing Edge Panel | | 7050-T7451 machined plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined areas. direction controlled part) | |
| [4] | Terminal Fitting | | 7050-T74 machined forging as given in BMS 7-214. Refer to the production drawing for the machined areas. direction controlled part) | |
| [5] | Backup Fitting, Main Landing Gear | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas. direction controlled part) | |
| [6] | Backup Fitting, Flap | | 7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas (Grain direction controlled part) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## ALLOWABLE DAMAGE 1 - OUTER WING FRONT SPAR FITTINGS

1. **Applicability**

   A. This subject gives the allowable damage limits for the outer wing front spar fittings shown in Location of the Outer Wing Front Spar, Figure 101/ALLOWABLE DAMAGE 1.




**Location of the Outer Wing Front Spar**
**Figure 101**

2. **General**

   A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 and Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limits.

   B. Remove the damage as necessary.

      (1) Refer to 51-10-02 for the inspection and removal of damage.

      (2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of all parts except the canted spar fitting.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked parts. Refer to 51-20-01.

(3) Apply a layer of BMS 10-20, Type II primer to the surfaces of the backup fittings, rib post fittings, and terminal fittings. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Apply a layer of BMS 10-11, Type I primer to the surfaces of the pitch load, actuator support, and the canted spar fittings. Refer to SOPM 20-41-02.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| **STRUCTURAL PART** | **PARAGRAPH** |
| End Pads | 4.A |
| Gussets | 4.A |
| Webs | 4.A |
| Attach Flanges | 4.A |
| Pitch Load Fitting Lugs | 4.B |
| Actuator Support Fitting Lugs | 4.C |

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-50-03 | Bearing and Bushing Replacement |

**4. Allowable Damage Limits**

A. End Pads, Gussets, and Attach Flanges

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E , F , and G .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Pitch Load Fitting Lugs

**NOTE**: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore to a maximum oversize diameter of 0.06 inch more than the initial bore diameter if:

- There is no damage on the edge of the lug
- You follow the bushing removal procedures as given in SOPM 20-50-03.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail A .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail A .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Actuator Support Fitting Lugs

**NOTE**: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore to a maximum oversize diameter of 0.06 inch more than the initial bore diameter if:

- There is no damage on the edge of the lug
- You follow the bushing removal procedures as given in SOPM 20-50-03.

(1) Cracks:

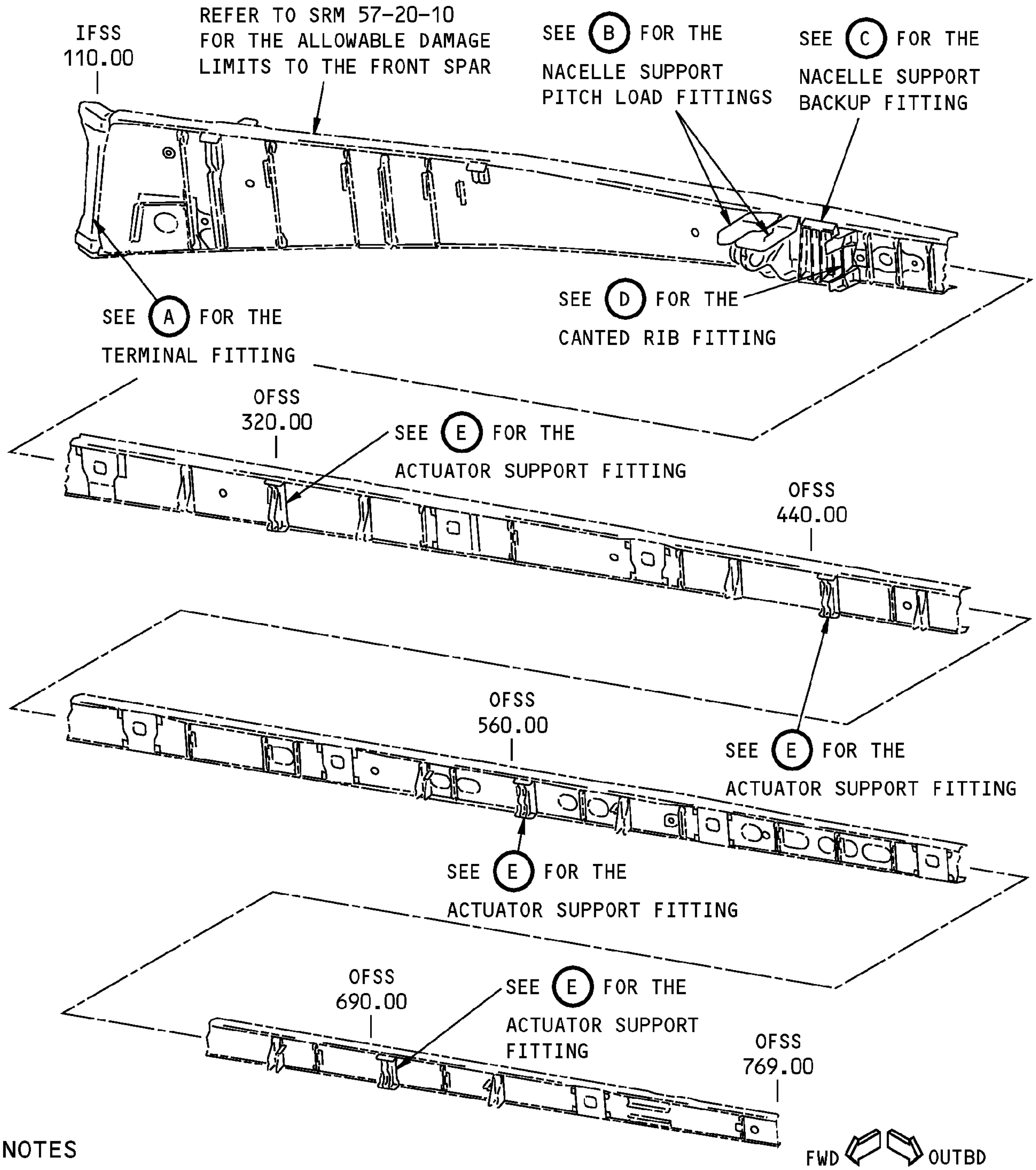
(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail B .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

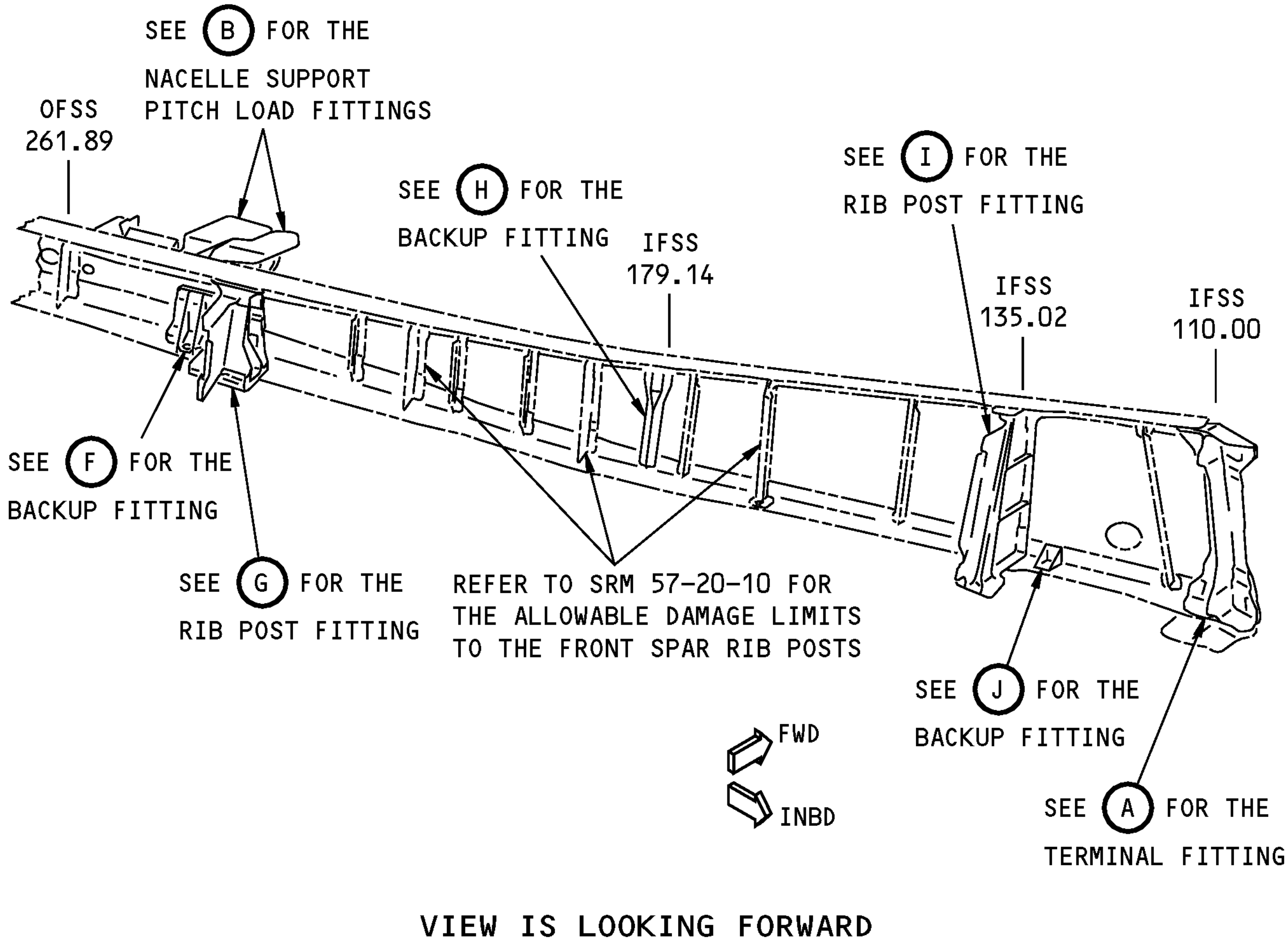



NOTES

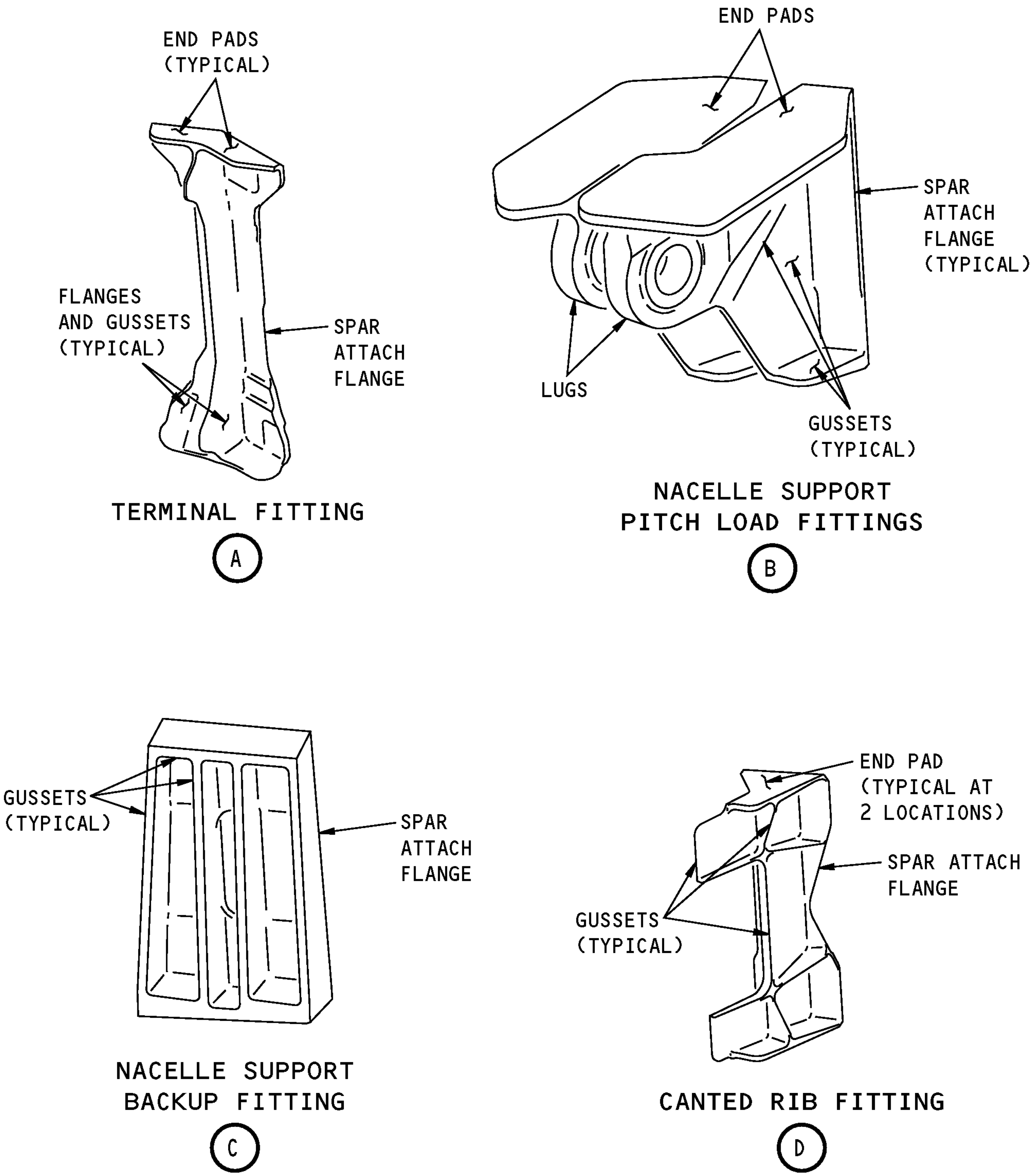
- ALL FITTINGS IDENTIFIED ARE MADE OF MACHINED ALUMINUM.

VIEW LOOKING AFT

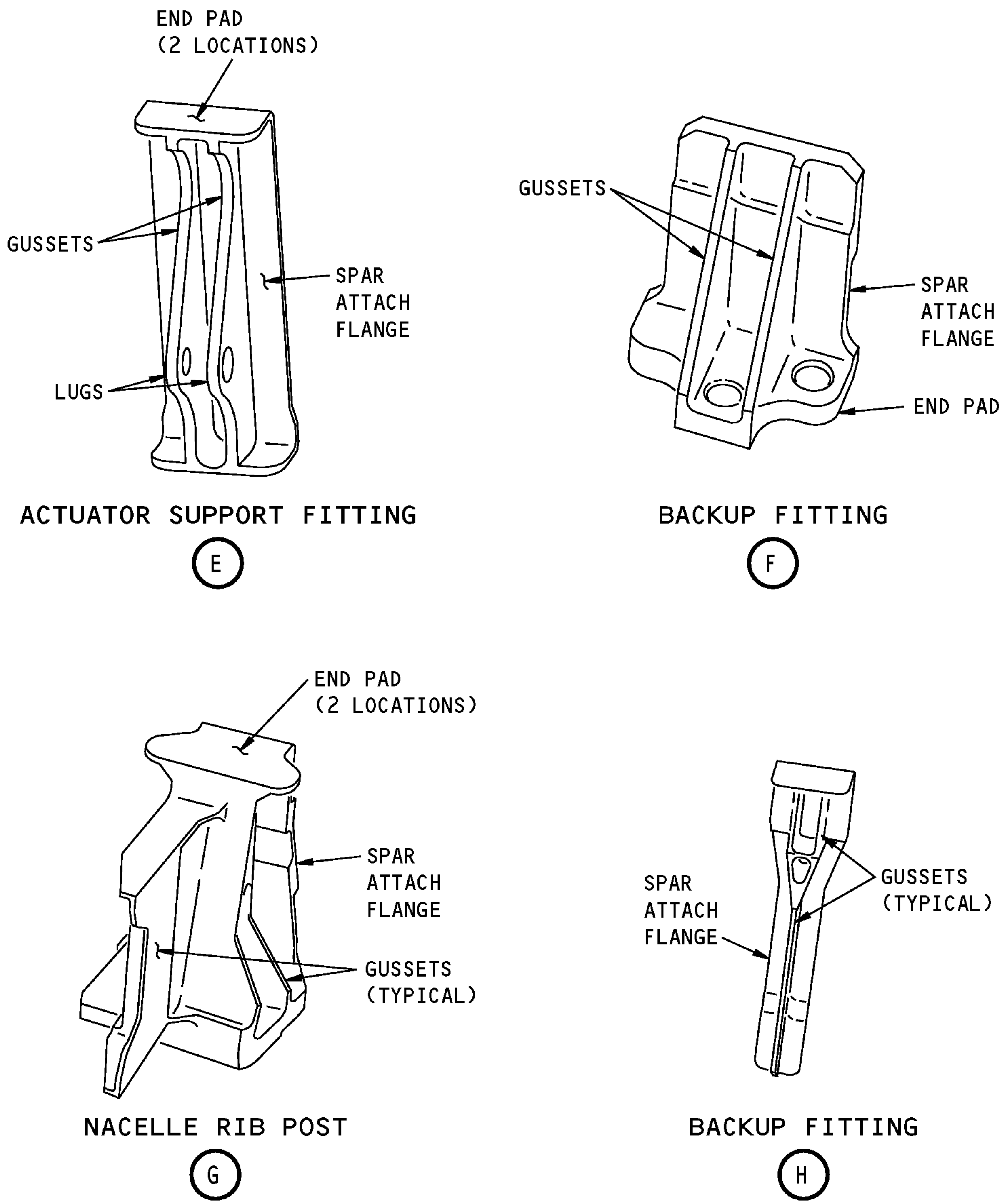
**Outer Wing Front Spar Fittings**
**Figure 102 (Sheet 1 of 5)**



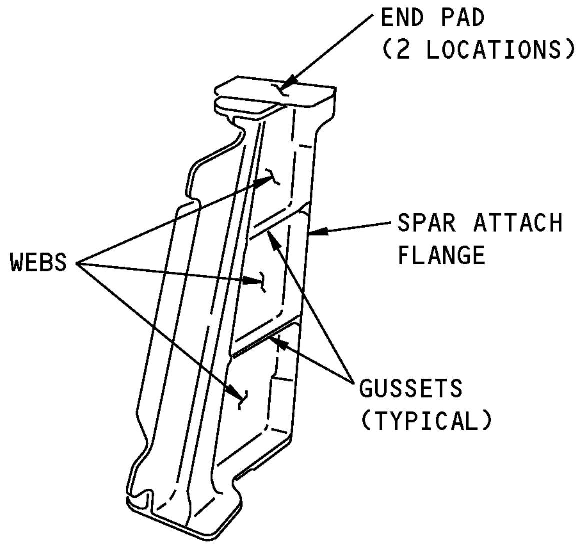

**Outer Wing Front Spar Fittings**
**Figure 102 (Sheet 2 of 5)**




**Outer Wing Front Spar Fittings**
**Figure 102 (Sheet 3 of 5)**

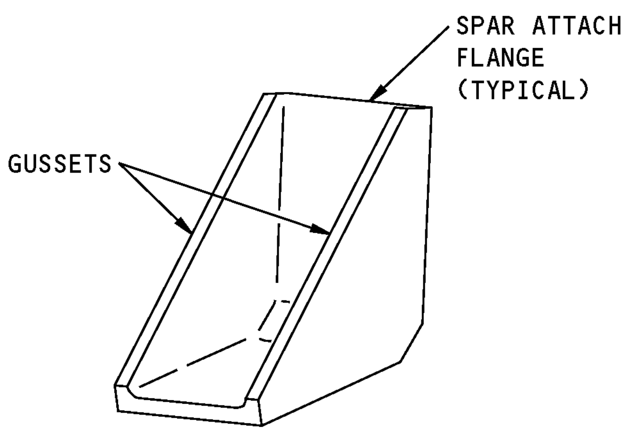



**Outer Wing Front Spar Fittings**
**Figure 102 (Sheet 4 of 5)**




RIB POST FITTING

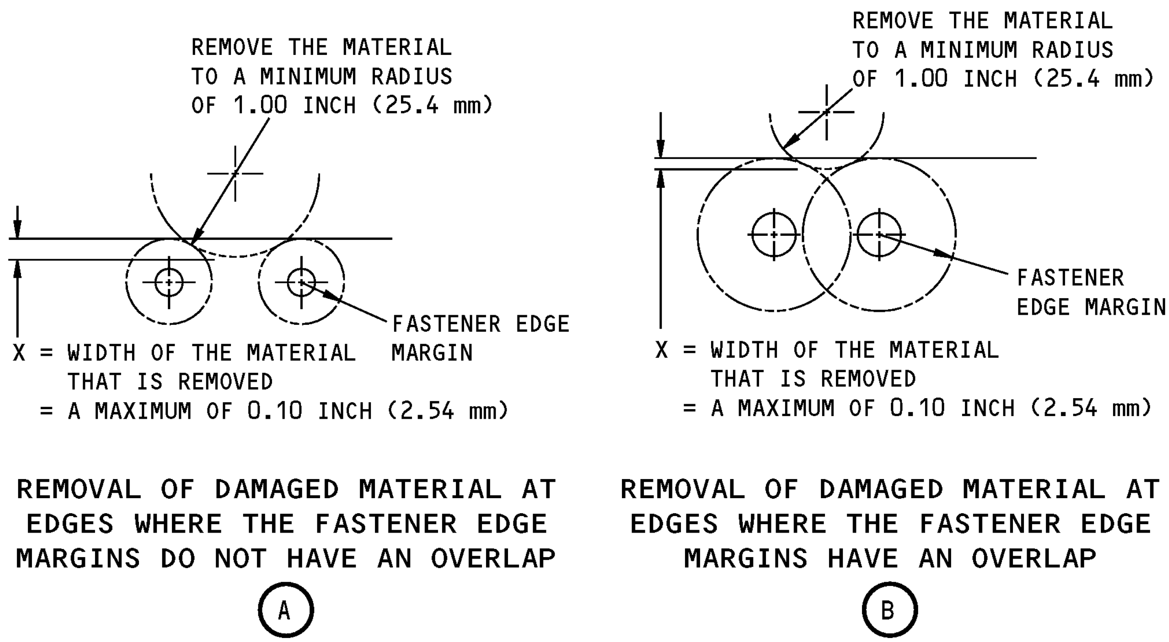
I



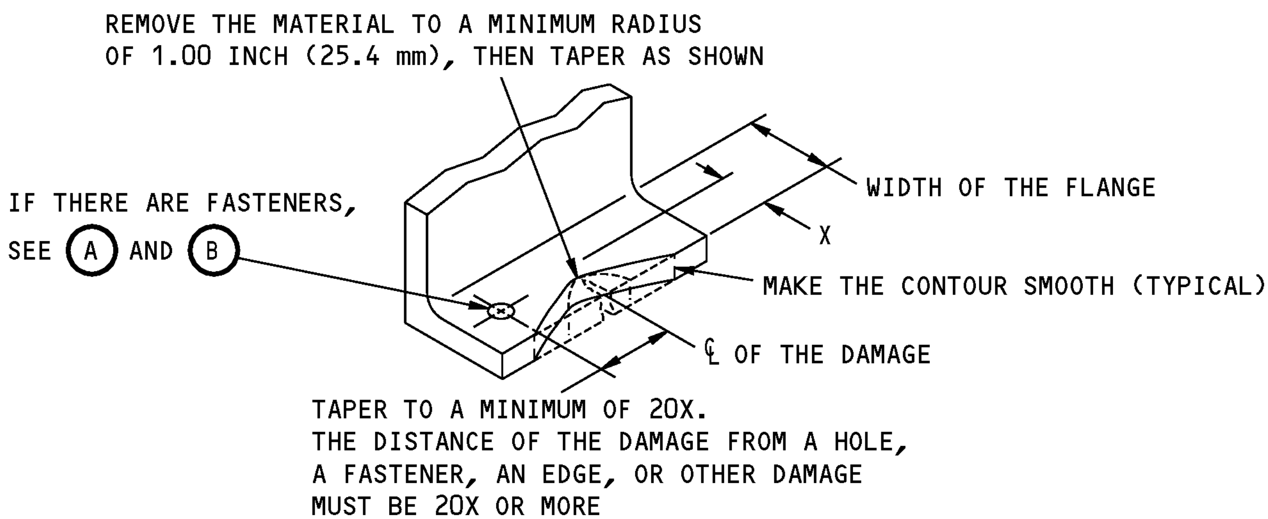

BACKUP FITTING

J

**Outer Wing Front Spar Fittings**
**Figure 102 (Sheet 5 of 5)**




**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 5)**

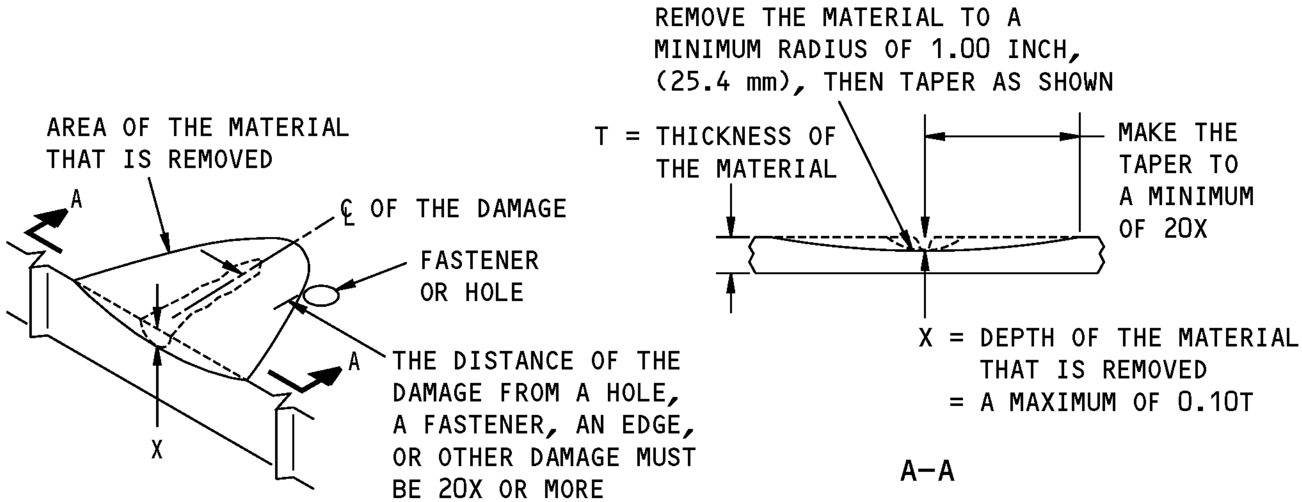



X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

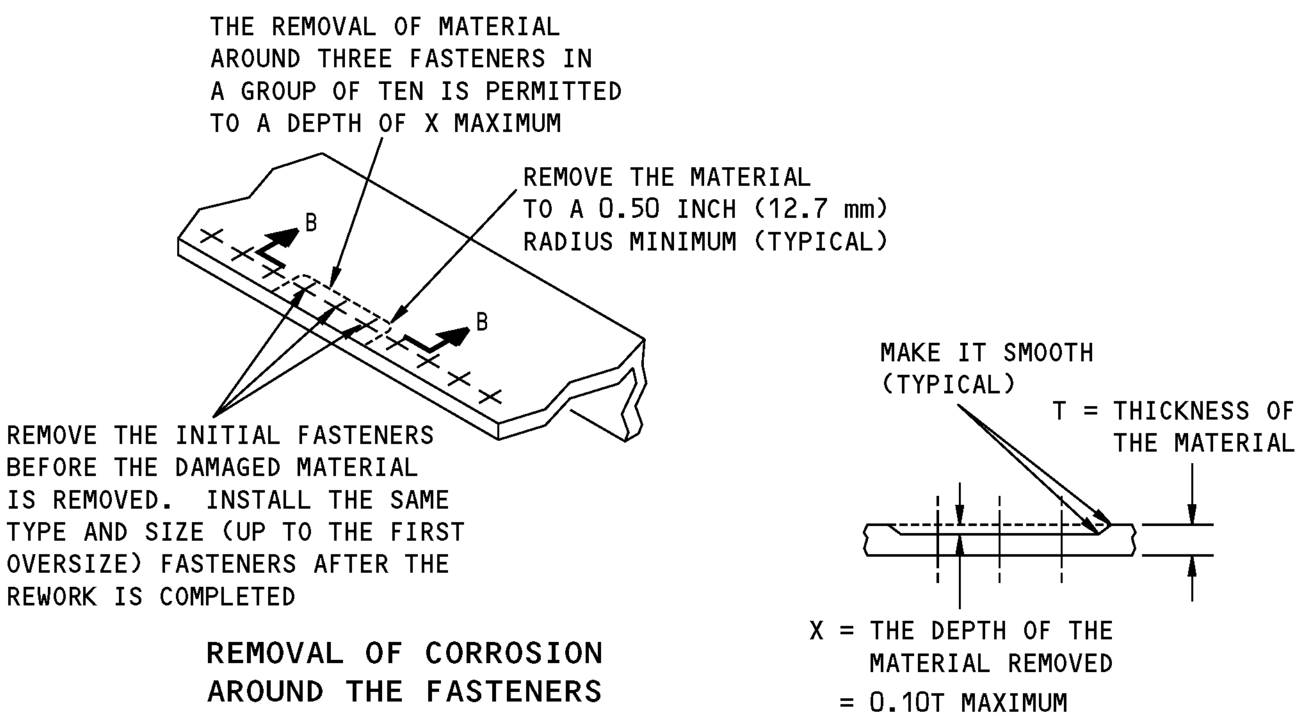
C

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 5)**




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

D

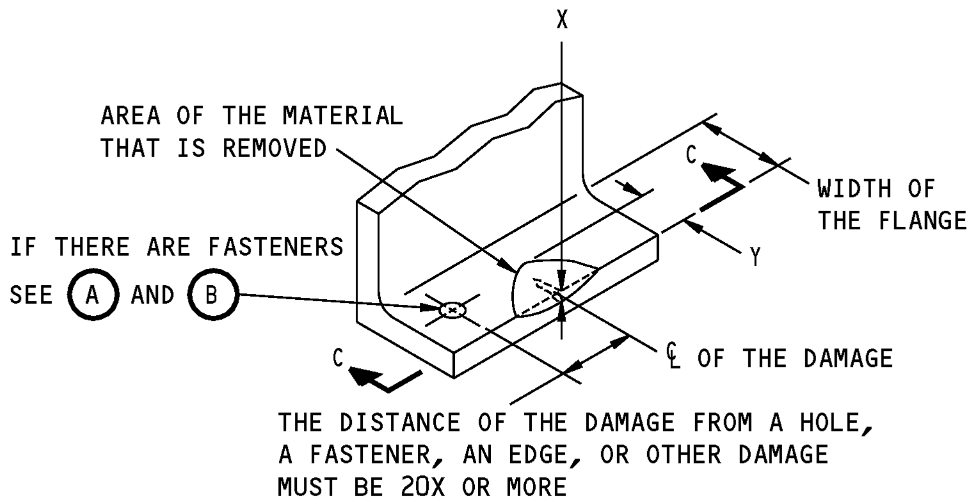



REMOVAL OF CORROSION
AROUND THE FASTENERS

E

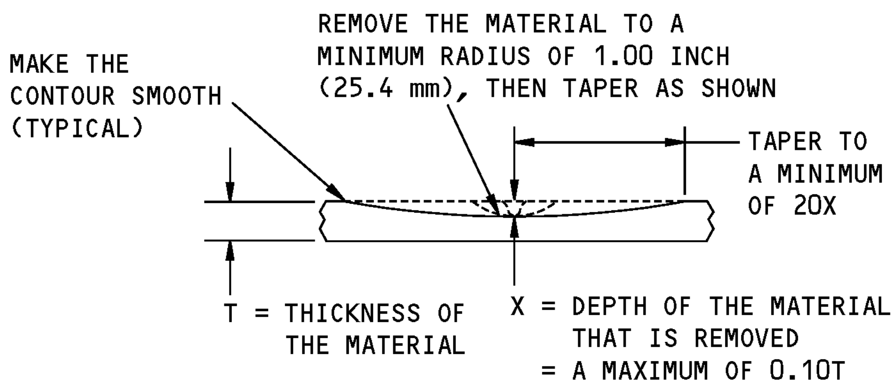
B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 5)**



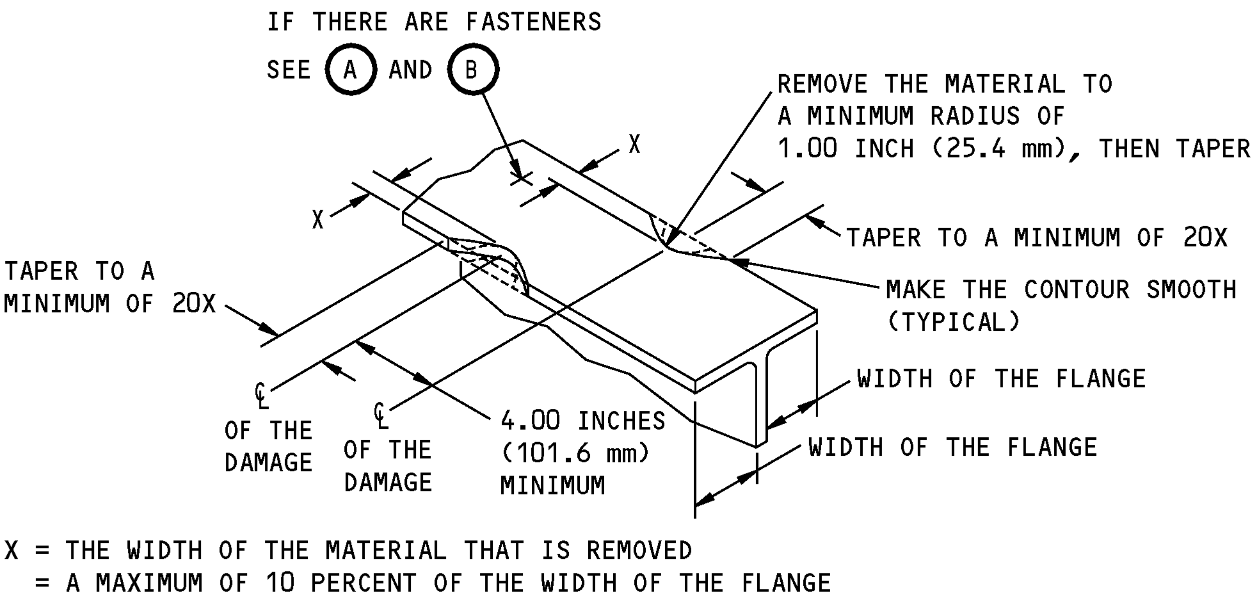

Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 1.0 INCH (25.4 mm)

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE
F



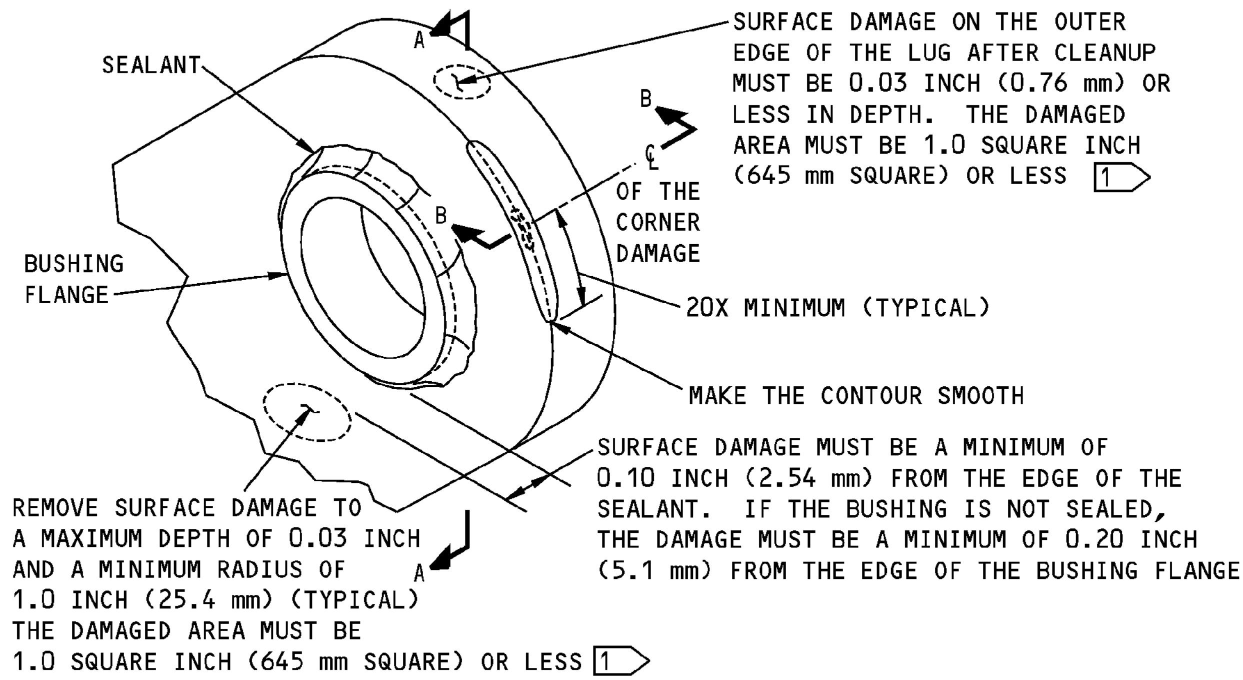

C-C

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 5)**




REMOVAL OF DAMAGED MATERIAL ON AN EDGE
G

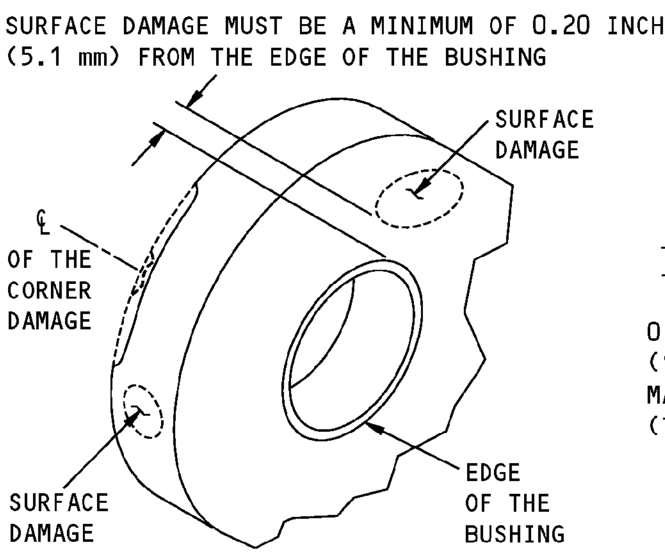
**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 5)**



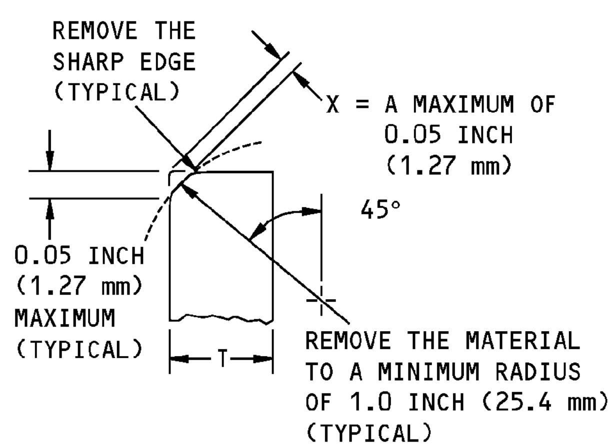

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

A

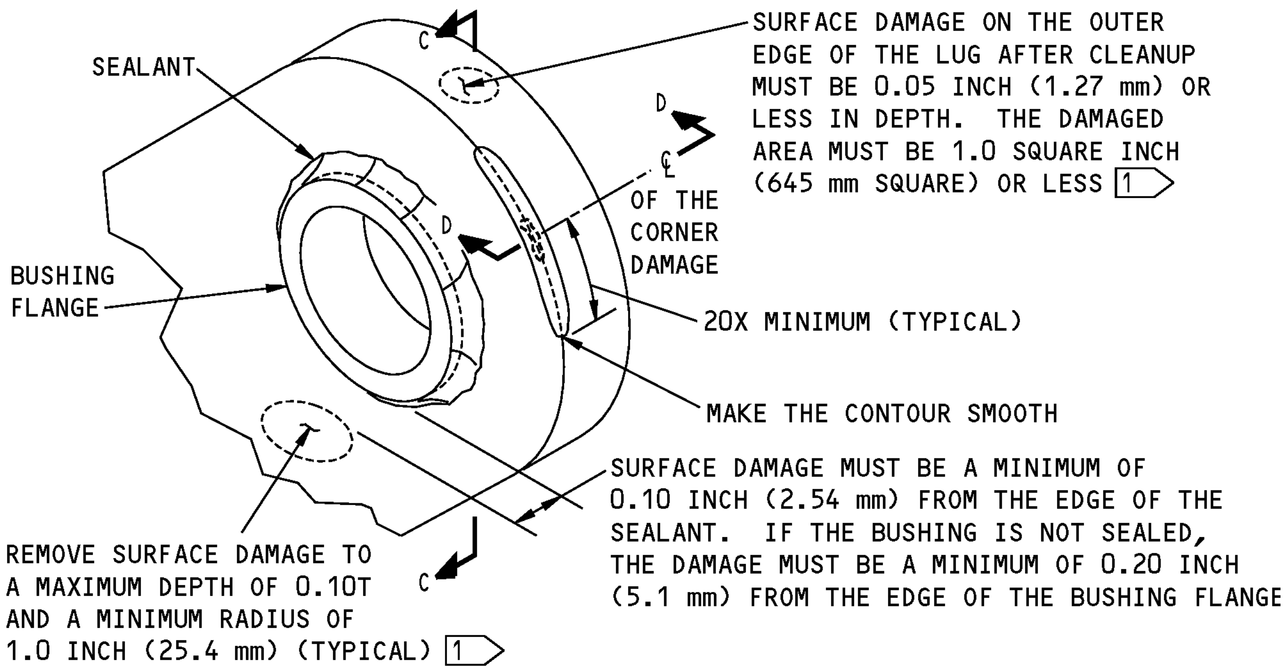



A-A




B-B 1

**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**

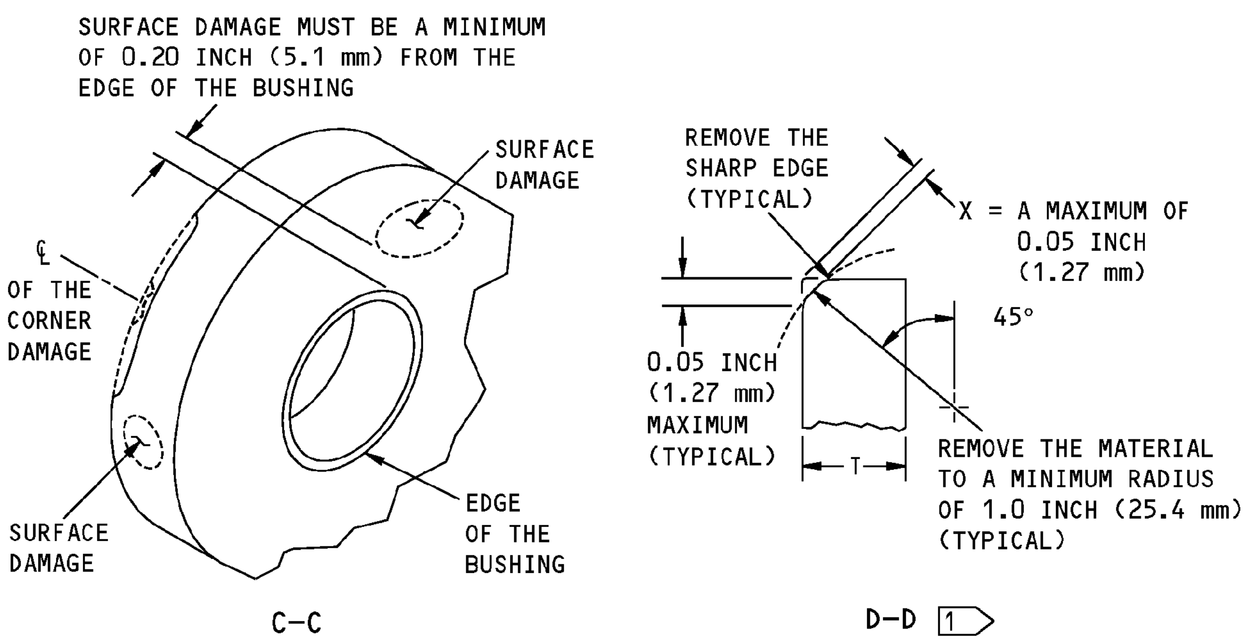



NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

B

**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**




NOTES

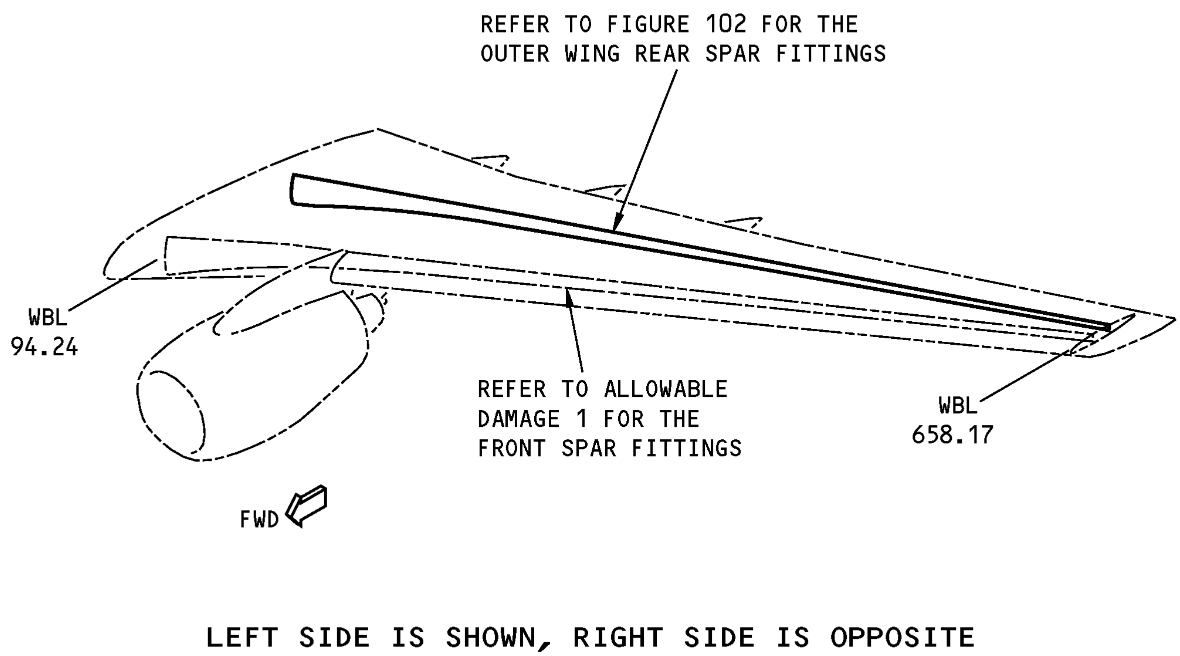
1 THESE ALLOWABLE DAMAGE LIMITS ARE APPLICABLE ONLY IF THERE IS NO DAMAGE IN THE BORE OF THE LUG.

**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**

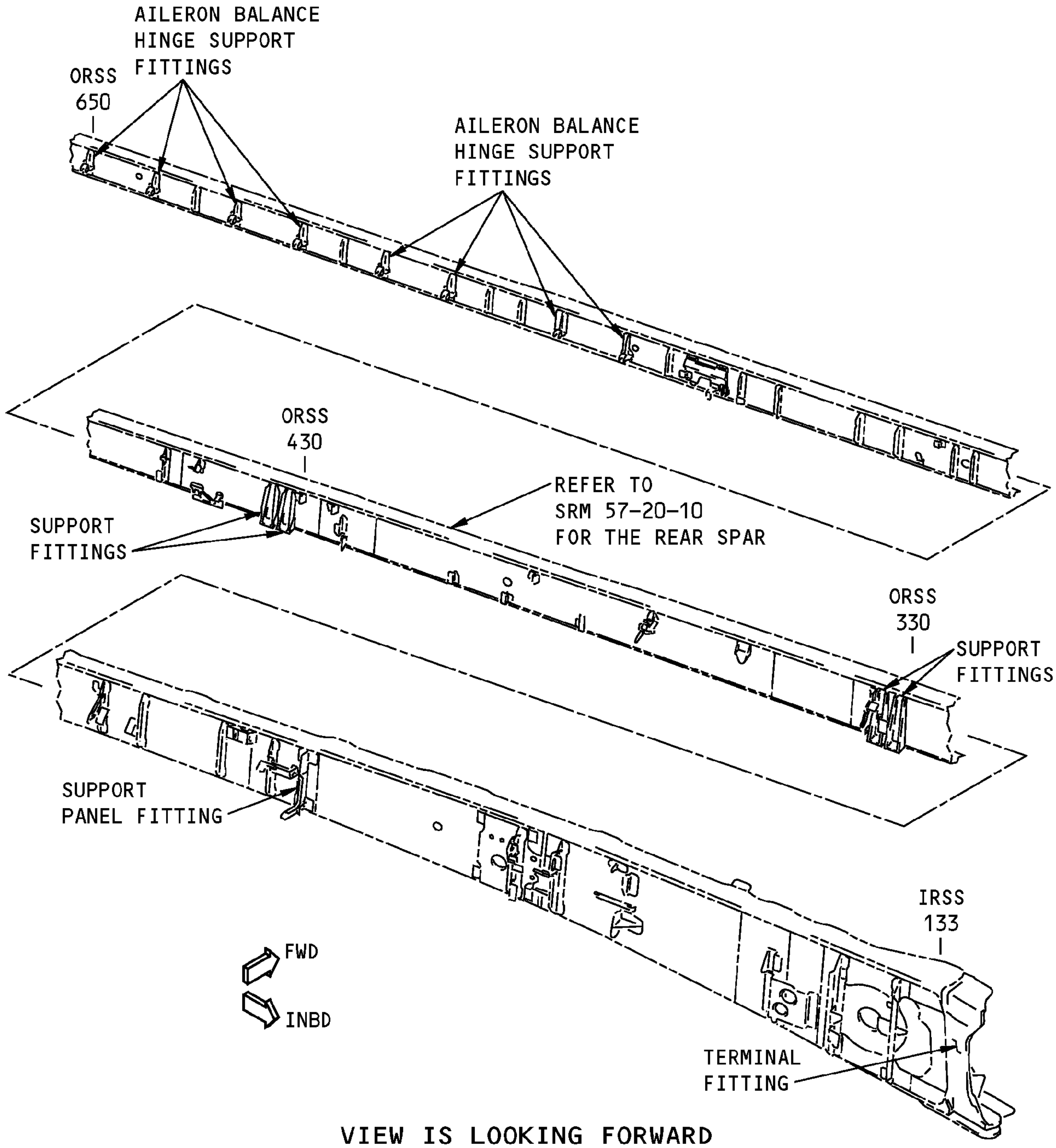
## ALLOWABLE DAMAGE 2 - OUTER WING REAR SPAR FITTINGS

### 1. Applicability

A. This subject gives the allowable damage limits for the outer wing rear spar fittings shown in Location of the Outer Wing Rear Spar Fittings, Figure 101/ALLOWABLE DAMAGE 2.




**Location of the Outer Wing Rear Spar Fittings**
**Figure 101**

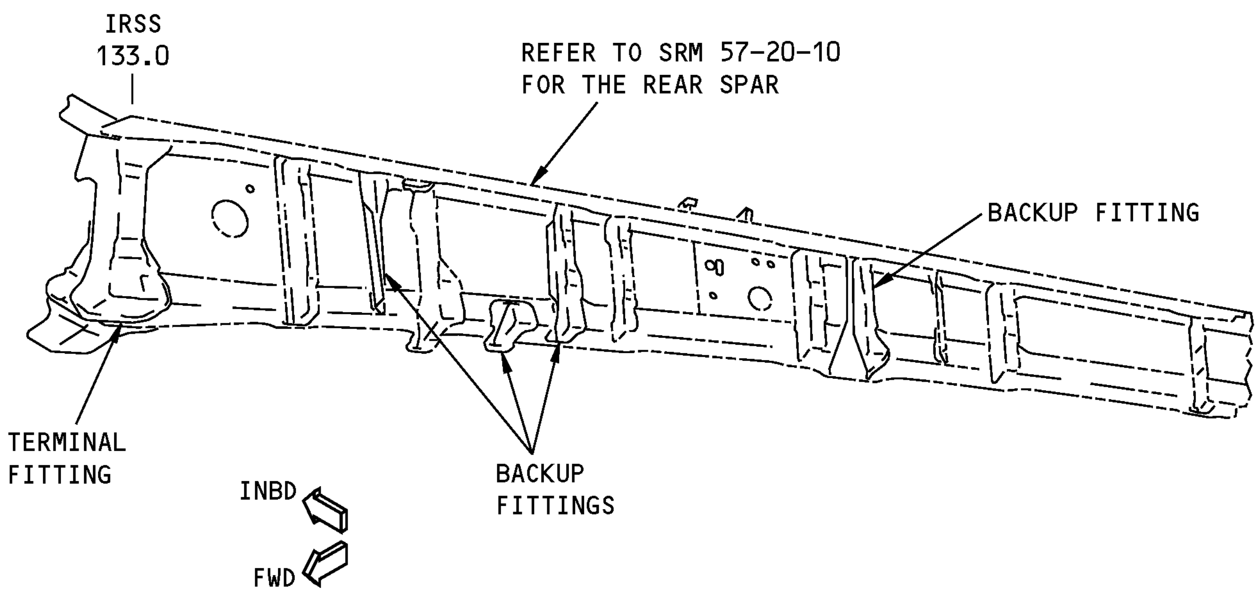



VIEW IS LOOKING FORWARD

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: ALL PARTS ARE MADE OF MACHINED ALUMINUM.

**Location of the Outer Wing Rear Spar Fittings**
**Figure 102 (Sheet 1 of 2)**




NOTE: ALL PARTS ARE MADE OF MACHINED ALUMINUM.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Location of the Outer Wing Rear Spar Fittings**
**Figure 102 (Sheet 2 of 2)**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of all parts except to the aileron balance hinge support fittings.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply a layer of BMS 10-20, Type II primer to the surfaces of the backup fittings. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Apply a layer of BMS 10-11, Type I primer to the surfaces of the support panel and aileron balance fittings. Refer to SOPM 20-41-02.

3. **References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-50-03 | Bearing and Bushing Replacement |

4. **Allowable Damage Limits**

A. Support Panel Fittings, Support Fittings, Terminal Fitting, and Backup Fittings

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , and H .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Actuator Balance Hinge Support Fittings

**NOTE**: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore to a maximum oversize diameter of 0.06 inch more than the initial bore diameter if:

- There is no damage on the edge of the lug
- You follow the bushing removal procedures as given in SOPM 20-50-03.

(1) Cracks:

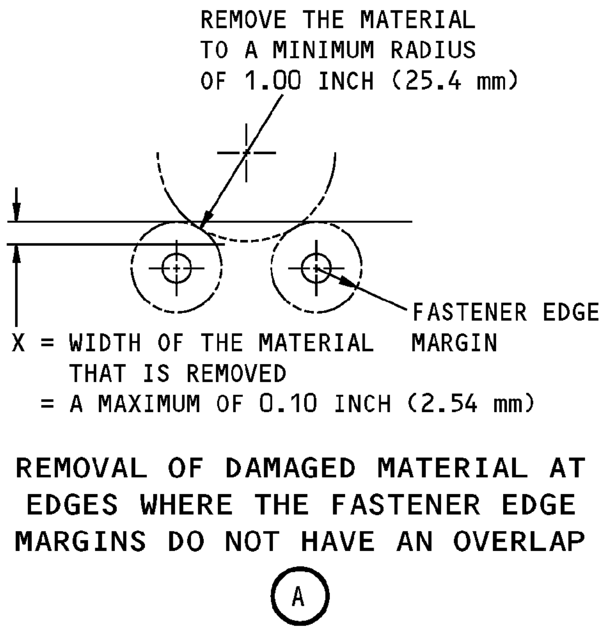
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

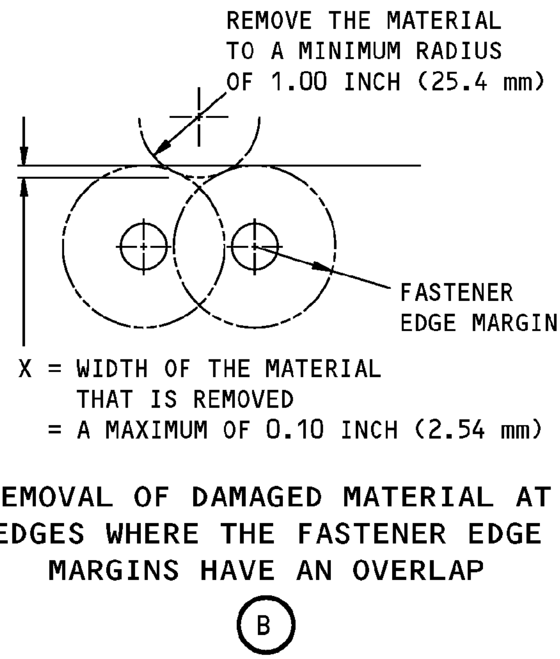
(2) Nicks, Gouges, Scratches, and Corrosion:

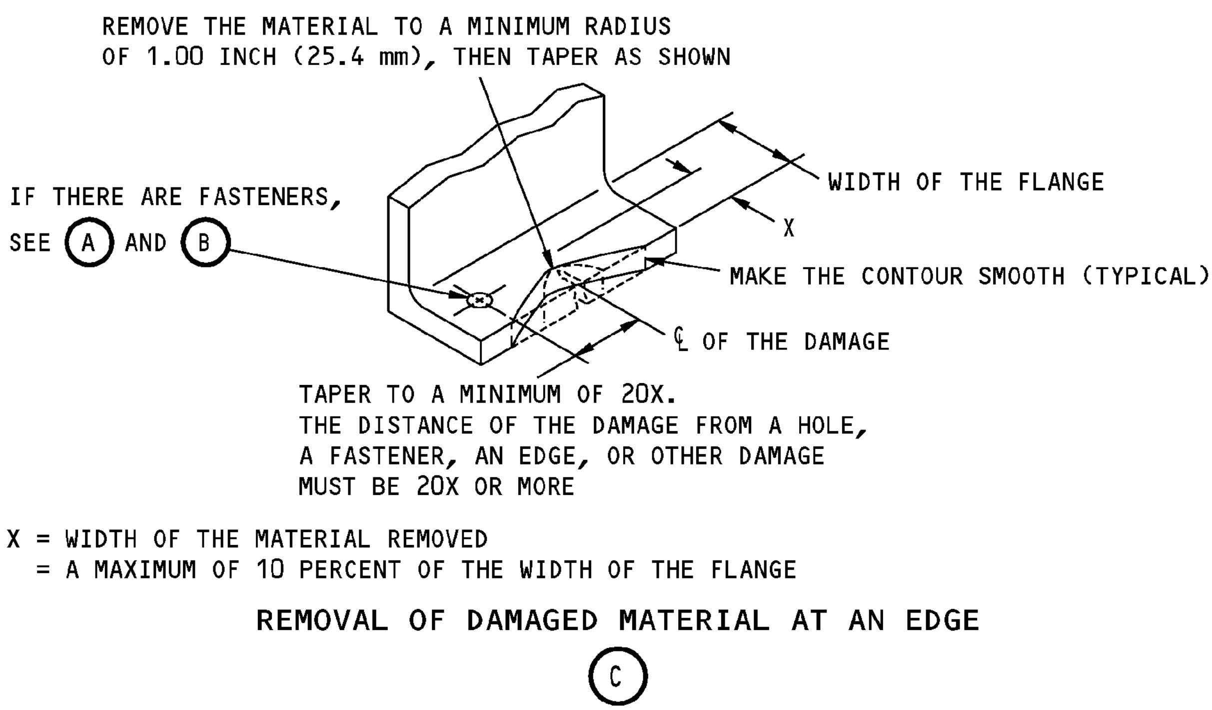
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F , G , and H .

(3) Dents are not permitted.

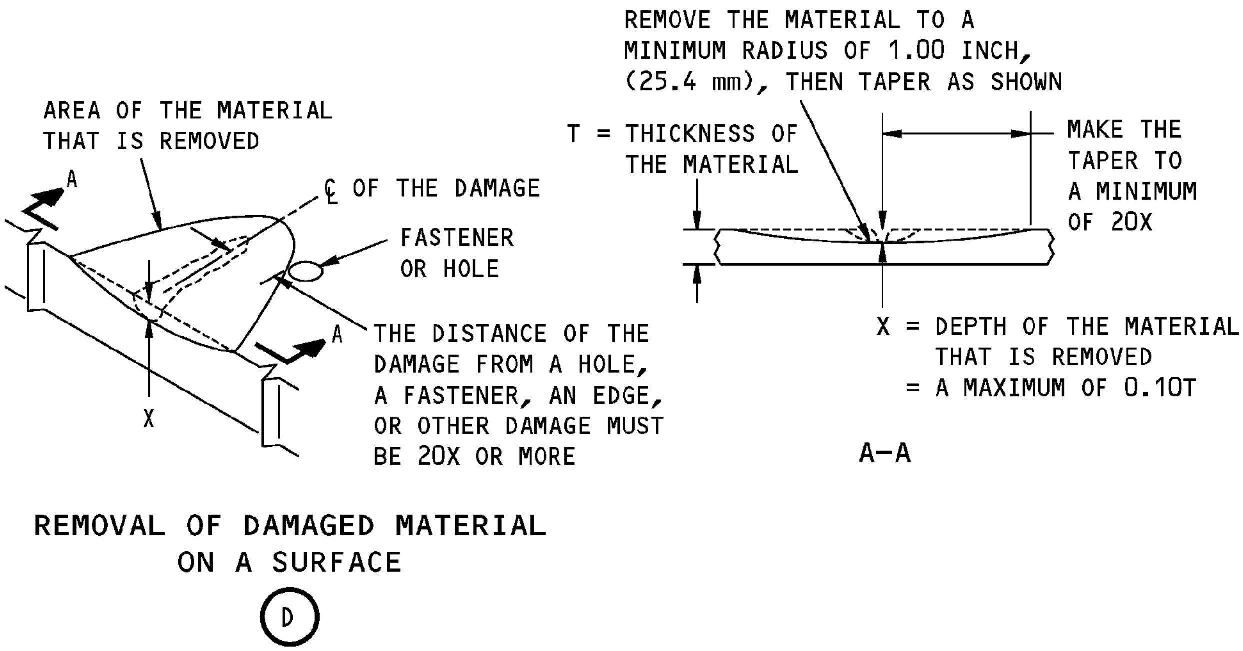
(4) Holes and Punctures are not permitted.

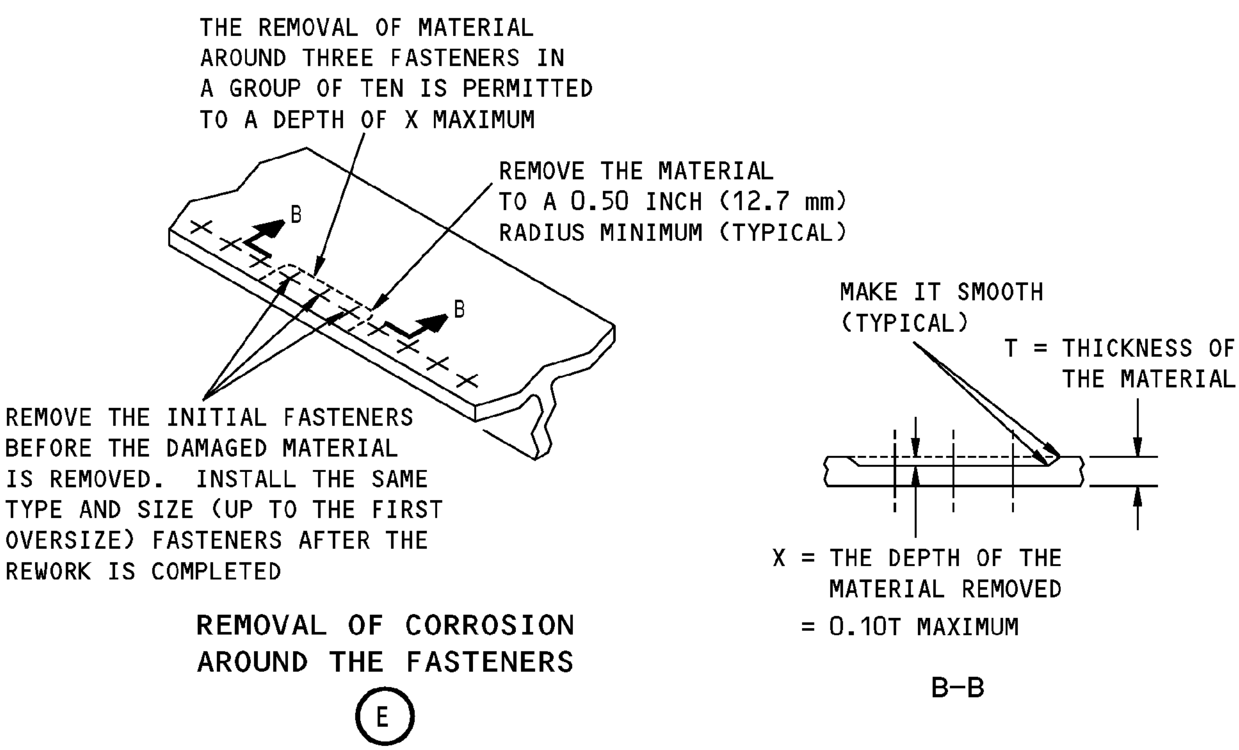


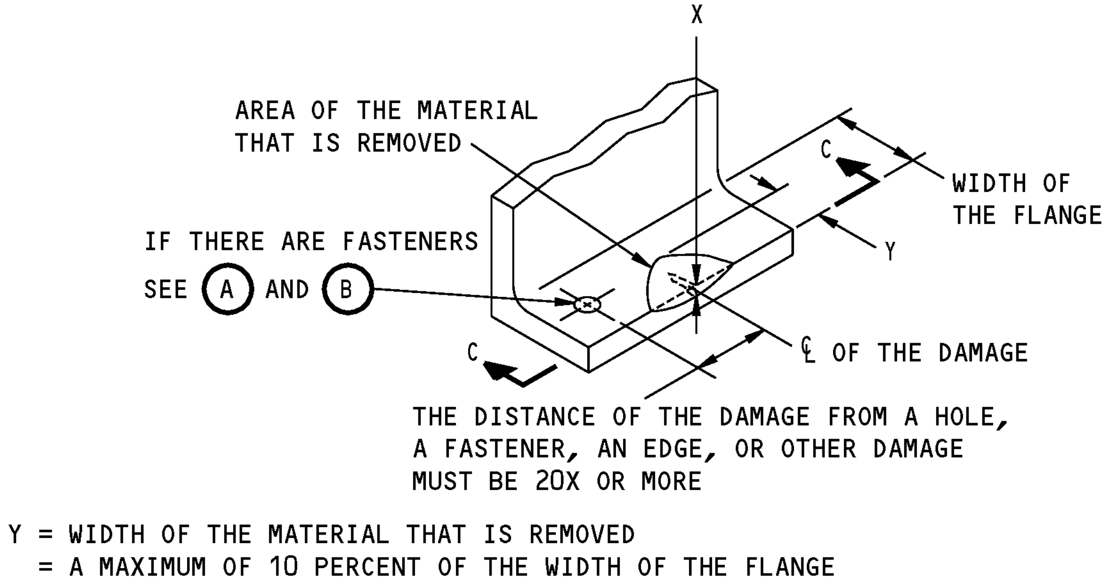




**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 5)**

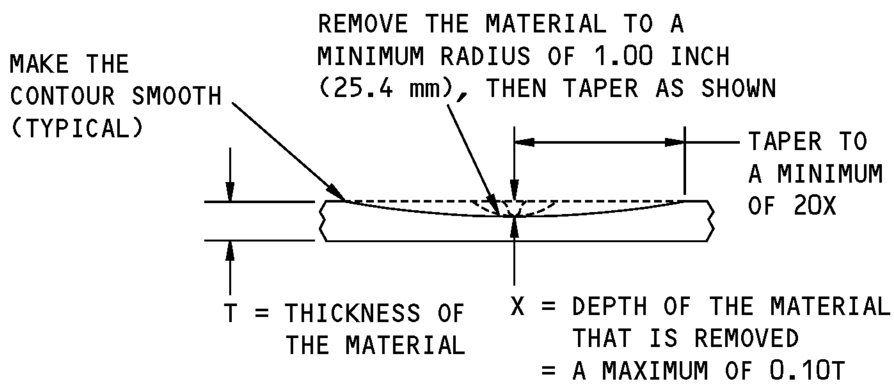



**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 5)**

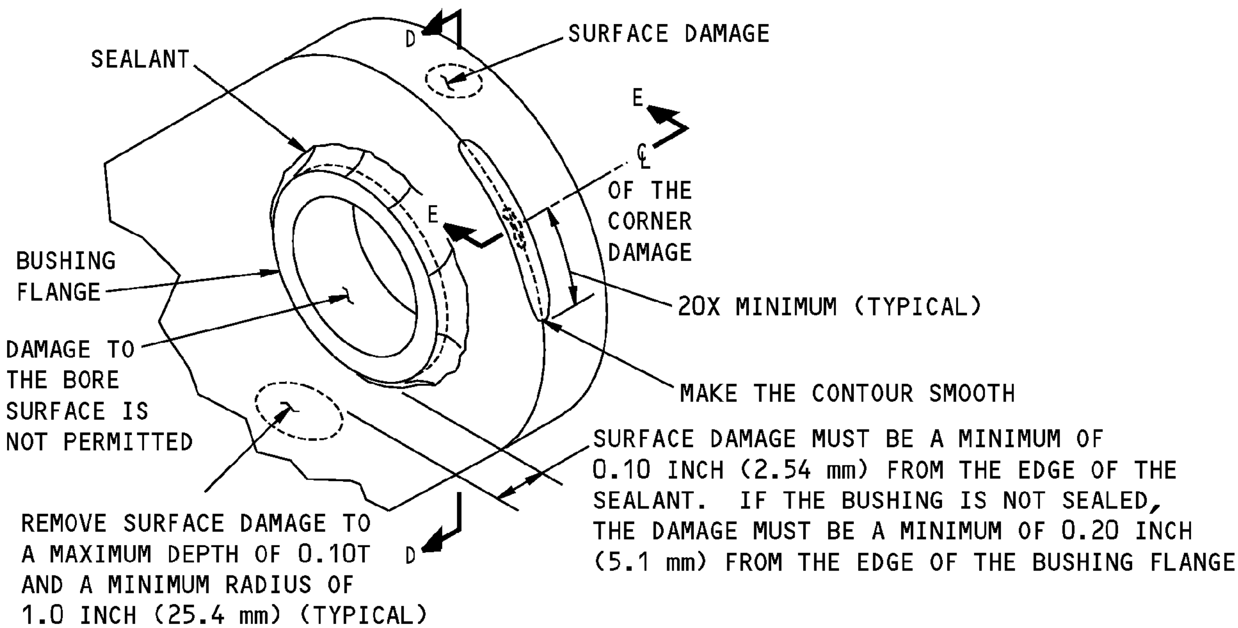



REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE
F




C-C

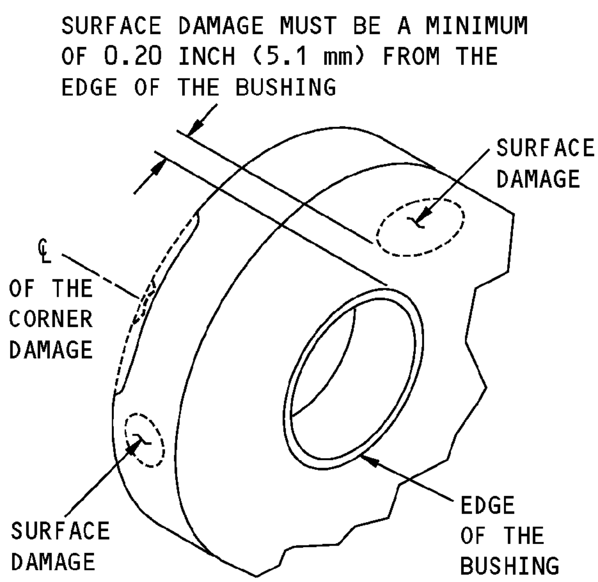
**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 5)**



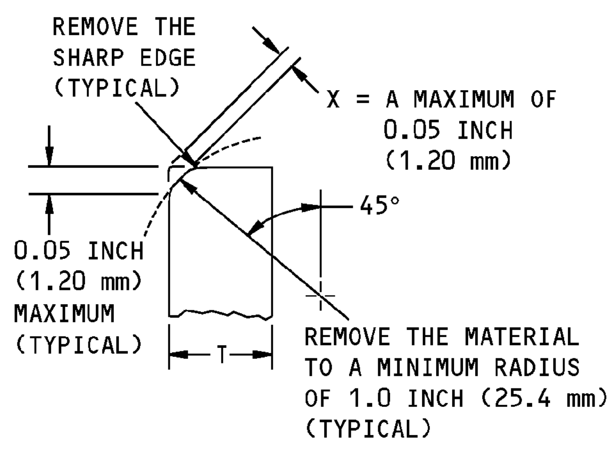

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

G

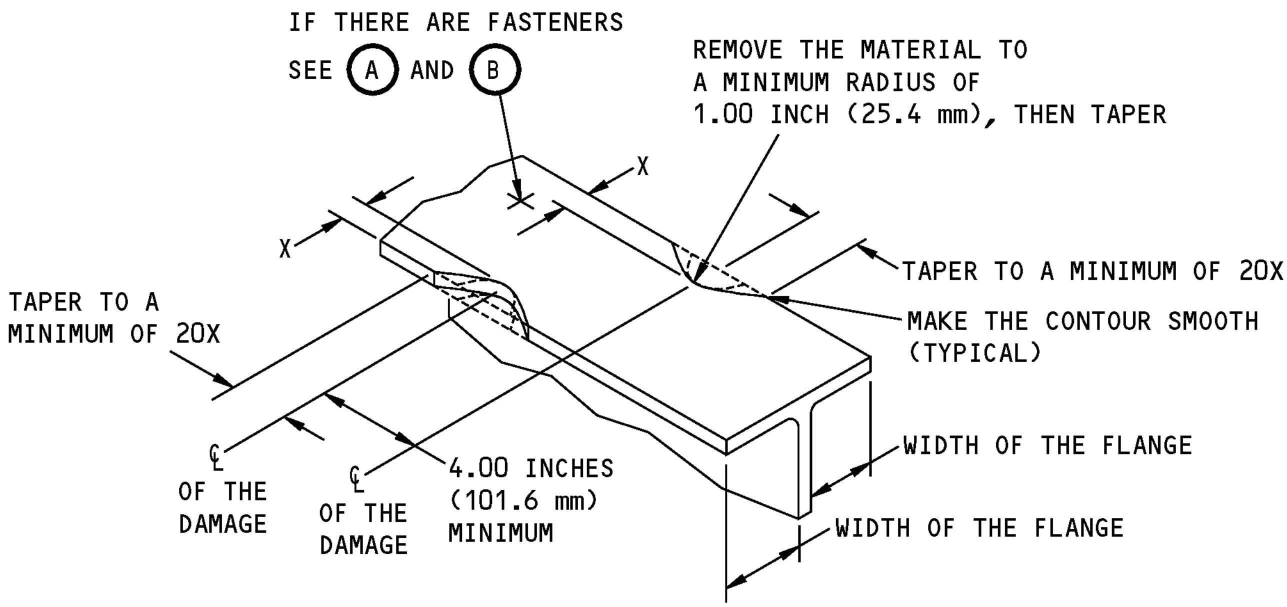



D-D




E-E

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 5)**



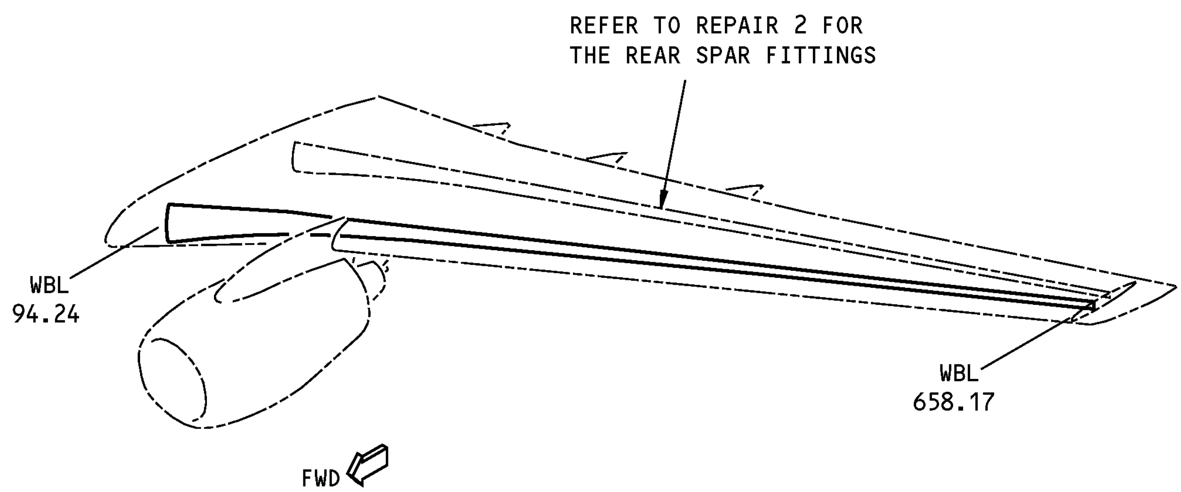

X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

H

**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 5)**

## REPAIR 1 - OUTER WING FRONT SPAR FITTINGS




NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Location of the Outer Wing Front Spar**
**Figure 201**

## REPAIR 2 - OUTER WING REAR SPAR FITTINGS - MAIN LANDING GEAR FORWARD TRUNNION SUPPORT FITTING CORROSION REPAIR

1. **Applicability**

   A. This repair is applicable to corrosion damage to the support fitting of the main landing gear forward trunnion. Refer to Figure 201/REPAIR 2 for the support fitting locations.

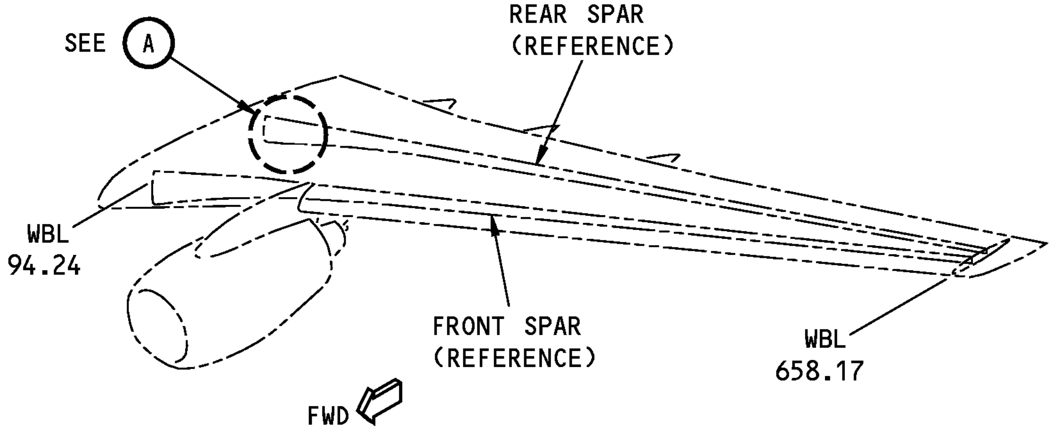
   B. This repair is applicable to corrosion damage on the surfaces of the support fitting shown in Figure 201/REPAIR 2.

   C. This repair is not applicable to crack damage to the support fitting. If you find a crack at a support fitting lug, replace the support fitting.

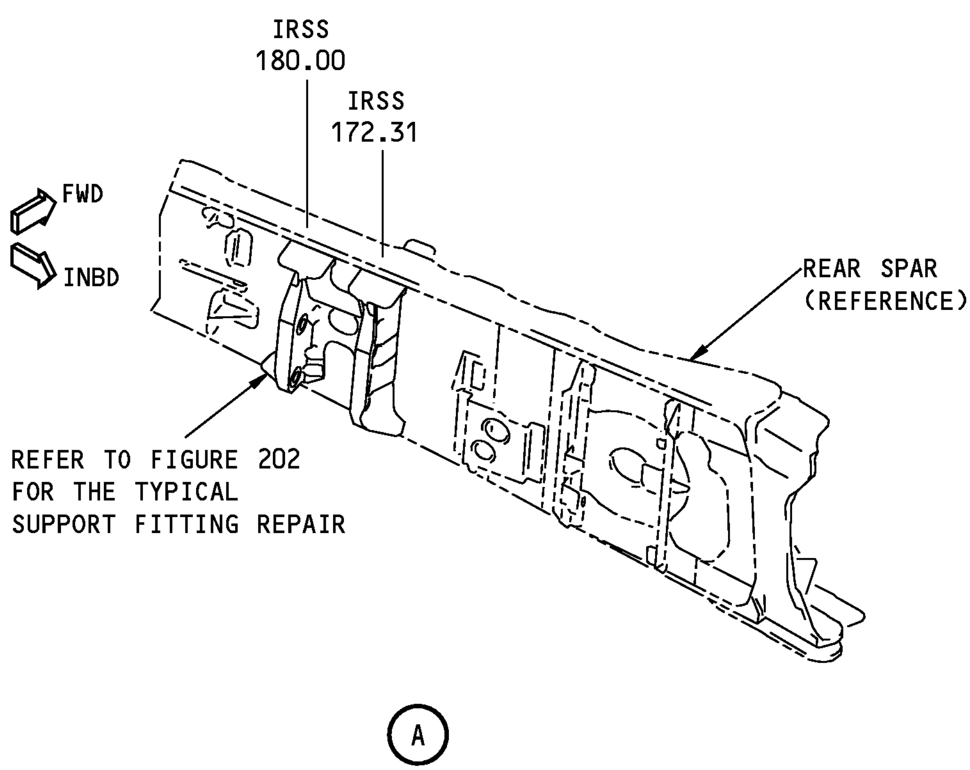
2. **General**

   A. This repair is a Category A damage tolerant repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 to find the definitions of the different categories of repairs.

   B. This repair uses the interference fit procedure to install the replacement bushings for a support fitting that is shown in Figure 201/REPAIR 2.




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE




**Main Landing Gear Forward Trunnion Support Fitting Location**
**Figure 201**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-20-13 | SURFACE ROUGHNESS FINISH REQUIREMENTS |
| AMM 32-11-83 P/B 401 | MAIN LANDING GEAR FORWARD TRUNNION BEARING ASSEMBLY - REMOVAL/INSTALLATION |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-30-03 | Standard Overhaul Practices Manual |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-42-05 | Bright Cadmium Plating |
| SOPM 20-50-03 | Bearing and Bushing Replacement |
| 737 NDT Part 6, 51-00-00, Figure 1 | Fastener Holes in Aluminum Parts (Meter Display) |
| 737 NDT Part 6, 51-00-00, Figure 4 | Surface Inspection of Aluminum Parts |

## 4. Repair Instructions

A. Get access to the damaged area, as necessary.

B. Remove the main landing gear forward trunnion bearing assembly as given in AMM PAGEBLOCK 32-11-83/401.

**CAUTION**: USE CARE WHEN YOU REMOVE THE BUSHINGS. THESE BUSHINGS WERE INSTALLED WITH AN INTERFERENCE FIT. IF YOU DO NOT, DAMAGE TO THE FITTING CAN OCCUR.

C. Remove the bushings from the support fitting where the corrosion damage occurs. Keep the removed bushings, if they are in satisfactory condition.

D. Do an inspection of the damaged area. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02 for the inspection procedure.

**CAUTION**: USE CARE WHEN YOU WORK ON THE SURFACE OF THE FITTING HOLE. SOME TOOLS AND REPAIR PROCEDURES CAN CAUSE DAMAGE TO THE FITTING HOLE. THE RESULT CAN BE A HOLE THAT IS LARGER THAN THE MAXIMUM DIMENSION THAT IS PERMITTED.

E. If there is corrosion damage on the outer surface of the support fitting around the bore, Remove the damaged material to a maximum depth of 0.150 in. (3.8 mm). Make the surface flat and parallel to the initial surface. Refer to Figure 202/REPAIR 2, Detail A, Detail B, and INSPECTION AND REMOVAL OF DAMAGE, 51-10-02

F. If there is corrosion damage on the inner surface of the bore(s) of the support fitting, do as follows:

(1) Get or make the tools to line ream the two adjacent support fitting bores as follows:

(a) Use a long piloted reamer to keep the bores coaxial to each other. The pilot part of the reamer must be non-cutting to prevent damage to the bore.

(b) Use a guide bushing in the adjacent bore to support the reamer, and protect the surfaces of the bore.

(2) Line ream and hone the bore(s) to remove the damaged material. Remove the damaged material up to a maximum bore diameter of 1.620 in. (41.15 mm). Refer to Figure 202/REPAIR 2, Detail A and INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

(a) Make sure that the bores are coaxial to each other.

(b) Make the surface finish of the bores equal to 63 microinches (1.6 micrometers) Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

G. Remove the sharp edges and burrs from the cut surfaces.

**WARNING:** KEEP THE SOLVENTS AWAY FROM SOURCES OF HEAT, FIRE, OR SPARKS. HEAT OR SPARKS CAN CAUSE AN EXPLOSION. DO NOT LET THE SOLVENTS GET INTO YOUR EYES OR ON YOUR SKIN OR CLOTHING. USE EYE PROTECTION. MAKE SURE THAT YOU USE AIRFLOW EQUIPMENT WHEN YOU DO WORK IN A CLOSED SPACE. IF YOU DO NOT OBEY, YOU CAN CAUSE INJURY TO PERSONS.

H. Clean the reworked areas of the support fitting with solvent. Refer to SOPM 20-30-03 for applicable solvents and procedures.

I. Do a penetrant inspection of the reworked areas to make sure that there is no other damage. Refer to SOPM 20-20-02. If you find more damage, do steps E through G, Paragraph 4./REPAIR 2 again.

**NOTE**: As an alternative, you can do a High Frequency Eddy Current (HFEC) inspection of the reworked area. Refer to 737 NDT Part 6, 51-00-00, Figure 1 and 737 NDT Part 6, 51-00-00, Figure 4.

J. Shot peen the outer surfaces of the support fitting where the material was removed, but do not apply the shot peen procedure to the bores of the support fitting.

(1) Keep 0.010A shot peening intensity and 2.0 coverage (manual procedure).

(2) Refer to SHOT PEENING, 51-20-06 for the shot number.

(3) Refer to SOPM 20-10-03 for shot peen procedure.

K. Make the surface finish of the shot peened surfaces of the support fitting equal to 125 microinches (3.2 micrometers) Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

L. Apply a chemical conversion coating to the bare surfaces of the support fitting. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

M. Apply a layer of BMS 10-11, Type I primer to the bare surfaces of the support fitting, but do not apply primer to the surfaces of the bores. Refer to SOPM 20-41-02.

N. Make the repair parts. Refer to Table 201/REPAIR 2 and Figure 203/REPAIR 2.

(1) Make or get the part [1] bushing with one of the two alternatives that follow:

(a) If you do not ream the support fitting bore, make part [1] bushing. Refer to Paragraph 5.A./REPAIR 2 for the necessary outer diameter dimension. Also, you can use the initial bushing that you removed, if it is not damaged, or get and use part number 115A5230-2 bushing from The Boeing Company for the part [1] bushing. Make sure that the diameter of the support fitting bore is 1.5602 in. (39.629 mm) to 1.5610 in. (39.649 mm). Refer to Paragraph 5.C./REPAIR 2 to calculate the necessary length of the part [1] bushing.

(b) If you ream the support fitting bore , make the part [1] bushing as shown in Figure 203/REPAIR 2, Detail A. Refer to Paragraph 5.B./REPAIR 2 to calculate the necessary outer diameter of the part [1] bushing. Refer to Paragraph 5.C./REPAIR 2 to calculate the necessary length of the part [1] bushing.

(2) Make the part [2] and the part [3] washers to replace the initial washers. Make the part [2] washer as shown in Figure 203/REPAIR 2, DETAIL C, and the part [3] washer as shown in Figure 203/REPAIR 2, DETAIL D. Refer to Paragraph 5.C.(2)/REPAIR 2 for the part [2] washer, and Paragraph 5.C.(3)/REPAIR 2 for the part [3] washer to calculate the necessary washer thickness.

**Table 201**: Repair Material

| ITEM | PART | QUANTITY | MATERIAL |
|---|---|---|---|
| [1] | Bushing | 4 | Use copper-beryllium (Cu – Be) tubing as given in AMS 4535. As an alternative, you can use copper-beryllium bar as given in AMS 4533. Refer to Figure 203/REPAIR 2, Detail A for the necessary dimensions. |
| [2] | Washer for the Fuse Pin Head Side. | 1 | Use 15-5PH CRES with a heat treat of 150 to 170 KSI as given in AMS 5659. Refer to Figure 203/REPAIR 2, Detail C for the necessary dimensions. |
| [3] | Washer for the Fuse Pin Nut Side | 1 | Use 15-5PH CRES with a heat treat of 150 to 170 KSI as given in AMS 5659. Refer to Figure 203/REPAIR 2, Detail D for the necessary dimensions. |

O. Apply the cadmium plating (Type II, Class 2) to the repair parts. Cadmium plating is optional to the surface of the inner diameter of the part [1] bushing. Make the thickness of cadmium plating from 0.0003 in. (0.0076 mm) to 0.0005 in. (0.0127 mm). Refer to the SOPM 20-42-05.

**WARNING:** USE PROTECTION WHEN YOU DO WORK WITH LIQUID NITROGEN. MAKE SURE THAT THE AREA HAS A GOOD FLOW OF CLEAN AIR. THE TEMPERATURE FOR LIQUID NITROGEN IS APPROXIMATELY MINUS 320°F (MINUS 196°C). IF YOU DO NOT OBEY, INJURY TO PERSONS CAN BE THE RESULT.

P. Use the shrink fit procedure to install the part [1] bushing with BMS 5-95 sealant as shown in Figure 202/REPAIR 2, Detail C. Use liquid nitrogen at -320°F (-196°C). Refer to SOPM 20-50-03.

(1) Make sure that the part [1] bushing is installed in the correct location.

(2) Hold the assembly in position until the assembly is at room temperature.

Q. Get or make the following tools to line ream the bushing through the pair of bushing holes to keep the bushing holes coaxial.

(1) Use a long piloted reamer to keep the bushing holes coaxial. The pilot part of the reamer must be non-cutting to prevent damage to the bushing.

(2) Use a guide bushing to slip into the adjacent bushing to support the reamer, and protect the bushing holes.

R. Line ream and hone the holes of the part [1] bushings to the final inner diameter of 1.2905 in. (32.78 mm) to 1.2915 in. (32.80 mm).

(1) Make sure that the pairs of bushing holes are coaxial as given in Figure 202/REPAIR 2, Detail C, Section B-B. If necessary, hone the pair of the replacement bushings at the same time to keep them coaxial.

(2) Make sure the surface finish of the holes is equal to 32 microinches (0.8 micrometers) Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

S. If you install new bushings to the support fitting then:

(1) Machine the flange of the bushings to the dimension shown in Figure 202/REPAIR 2 Detail C, section B-B. Re-cad plating of the bushing is not necessary after you machine the bushing flange.

(2) Machine the chamfer of the bushing flange as shown in Figure 202/REPAIR 2 Detail C, section B-B.

T. Clean the bushing holes to make sure that all the honing debris and machining oil are removed. Refer to SOPM 20-30-03.

U. Install the main landing gear forward trunnion bearing assembly as given in AMM PAGEBLOCK 32-11-83/401, but:

(1) If you removed material from around the bores of the support fitting, do the procedure that follows:

(a) Install the part [2] repair washer with BMS 3-38 corrosion inhibiting compound at the head side of the fuse pin. The part [2] washer replaces the initial special washer. Discard the initial special washer. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01 and Figure 204/REPAIR 2. Make sure that the washer is concentric with the support fitting bore when you install the fuse pin.

(b) Install the part [3] repair washer with BMS 3-38 corrosion inhibiting compound at the nut side of the fuse pin. The part [3] washer replaces the initial special washer. Discard the initial special washer. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01 and Figure 204/REPAIR 2. Make sure that the washer is concentric with the support fitting bore when you install the fuse pin.

(c) Apply a fillet seal with BMS 5-95 sealant to the edges of the part [2] and part [3] washers. Refer to REPAIR SEALING, 51-20-05.

V. Install the components that you removed for access, as necessary.

**5. Repair Parts Dimensions**

A. If you do not ream the bore of the support fitting, make the diameter of the bore 1.5602 in. (39.629 mm) to 1.5610 in. (39.649 mm). Make the outer diameter of the part [1] bushing 1.5626 in. (39.690 mm) to 1.5638 in. (39.721 mm) for an interference fit.

**NOTE**: As an alternative, you can get and use a part number 115A5230-2 bushing from The Boeing Company for the part [1] bushing, or you can use the initial bushing that you removed, if it is in satisfactory condition.

B. If you ream the bore of the support fitting, make the outer diameter of the part [1] bushing 0.0016 in. (0.041 mm) to 0.0036 in. (0.091 mm) larger for an interference fit. Refer to Figure 203/REPAIR 2, Detail A to identify the outer diameter (D6) of the part [1] bushing. Use the steps that follow to calculate the outer diameter of the part [1] bushing:

(1) After you remove the damage, measure the diameter of the bore of the support fitting. Make a record of the minimum and maximum value of the diameter of the bore. Make sure that the maximum diameter of the bore is not larger than 1.620 in. (41.15 mm).

(2) Calculate the sum of the minimum diameter of bore and 0.0036 in. (0.091 mm). Use the result value as a maximum outer diameter of the part [1] bushing.

(3) Calculate the sum of the maximum diameter of bore and 0.0016 in. (0.041 mm). Use the result value as a minimum outer diameter of the part [1] bushing.

(4) Example: Assume that you ream the bore of the support fitting to a diameter of 1.6000 in. (40.640 mm) to 1.6008 in. (40.660 mm):

(a) The maximum outer diameter of the part [1] bushing equals the sum of 1.6000 in. (40.640 mm) and 0.0036 in. (0.091 mm) = 1.6036 in. (40.731 mm).

(b) The minimum outer diameter of the part [1] bushing equals the sum of 1.6008 in. (40.660 mm) and 0.0016 in. (0.041 mm) = 1.6024 in. (40.701 mm).

(c) The outer diameter of the part [1] bushing must be 1.6024 in. (40.701 mm) to 1.6036 in. (40.731 mm).

C. If you removed the damage from the outer surface of the support fitting around the bore:

(1) Make the part [1] bushing with a length that is shorter than the length of the initial bushing. Refer to Figure 203/REPAIR 2, Detail A to identify the length (L1) of the part [1] bushing. Use the steps that follow to calculate the necessary length (L1) of the part [1] bushing:

(a) Measure the depth of the material that you removed from the outer surface of the support fitting around the bore. Refer to Figure 202/REPAIR 2, Section A-A to identify the depth (X) of the material that you removed. Make sure that the maximum depth of removed material is not larger than 0.150 in. (3.8 mm). Make a record of the minimum and maximum values of the removed material depth.

(b) Calculate the sum of the minimum and maximum values of removed material depth.

(c) Divide the sum of the minimum and maximum values of removed material depth by two. Use the result value as the average depth of removed material.

(d) Subtract the value of the average depth of removed material from 1.155 in. (29.34 mm). Use the result value as the nominal length of the part [1] bushing.

(e) Add 0.005 in. (0.127 mm) to the nominal length of the part [1] bushing. Use the result value as a maximum length of the part [1] bushing.

(f) Subtract 0.005 in. (0.127 mm) from the nominal length of the part [1] bushing. Use the result value as a minimum length of the part [1] bushing.

(g) Example: Assume that you remove a depth of 0.100 in. (2.54 mm) to 0.102 in. (2.59 mm) of material from the outer surface of the support fitting:

1) The sum of the minimum and maximum values of removed material depth is equal to 0.100 in. (2.54 mm) plus 0.102 in. (2.59 mm) = 0.202 in. (5.13 mm).

2) The average depth of removed material is equal to 0.202 in. (5.13 mm) divided by 2 = 0.101 in. (2.57 mm).

3) The nominal length of the part [1] bushing is equal to 1.155 in. (29.34 mm) minus 0.101 in. (2.57 mm) = 1.054 in. (26.77 mm).

4) The maximum length of the part [1] bushing is equal to the sum of 1.054 in. (26.772 mm) and 0.005 in. (0.127 mm) = 1.059 in. (26.899 mm).

5) The minimum length of the part [1] bushing is equal to 1.054 in. (26.772 mm) minus 0.005 in. (0.127 mm) = 1.049 in. (26.645 mm).

6) The length (L1) of the part [1] bushing must be 1.049 in. (26.645 mm) to 1.059 in. (26.899 mm).

(2) Make the part [2] washer with a thickness that is larger than the thickness of the initial special washer. Refer to Figure 203/REPAIR 2, Detail C and Figure 204/REPAIR 2 to identify the part [2] washer. Use the steps that follow to calculate the necessary thickness of the part [2] washer (T1):

(a) Measure the depth of the material that you removed from the outer surface of the support fitting around the bore. Refer to Figure Figure 202/REPAIR 2, Section A-A to identify the depth (X) of the material that you removed. Make sure that the maximum depth of removed material is not larger than 0.150 in. (3.810 mm). Make a record of the minimum and maximum values of the removed material depth.

(b) Calculate the sum of the minimum and maximum values of removed material depth.

(c) Divide the sum of the minimum and maximum values of removed material depth by two. Use the result value as the average depth of removed material.

(d) Calculate the sum of the average depth of removed material and 0.070 in. (1.778 mm). Use the result value as a maximum thickness of the part [2] washer.

(e) Calculate the sum of the average depth of removed material and 0.060 in. (1.524 mm). Use the result value as a minimum thickness of the parts [2] washer.

(f) Example: Assume that the material you removed from the outer surface of the support fitting is a depth of 0.100 in. (2.540 mm) to 0.102 in. (2.591 mm).

1) The sum of the minimum and maximum values of removed material depth is equal to 0.100 in. (2.540 mm) plus 0.102 in. (2.591 mm) = 0.202 in. (5.131 mm)

2) The average depth of removed material is equal to 0.202 in. (5.131 mm) divided by 2 = 0.101 in. (2.565 mm).

3) The maximum thickness of the part [2] washer is equal to the sum of 0.101 in. (2.565 mm) and 0.070 in. (1.778 mm) = 0.171 in. (4.343 mm)

4) The minimum thickness of the part [2] washer is equal to the sum of 0.101 in. (2.565 mm) and 0.060 in. (1.524 mm) = 0.161 in. (4.089 mm)

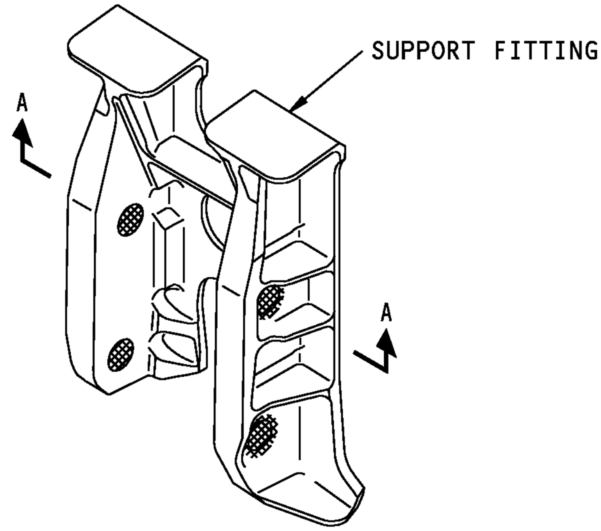
5) The thickness of the part [2] washer is 0.161 in. (4.089 mm) to 0.171 in. (4.343 mm)

(3) Make the part [3] washer with a thickness that is larger than the thickness of the initial special washer. Refer to Figure 203/REPAIR 2, Detail D and Figure 204/REPAIR 2 to identify the part [3] washer. Use the steps shown in Paragraph 5.C.(2)(a)/REPAIR 2 through Paragraph 5.C.(2)(f)/REPAIR 2 to calculate the necessary thickness of the part [3] washer (T1).

D. If there is no damage to be removed from the outer surface of the support fitting around the bore:

(1) Make the part [2] washer thickness the same as the initial special washer thickness. Refer to Figure 203/REPAIR 2, Detail C and Figure 204/REPAIR 2 to identify the part [2] washer.

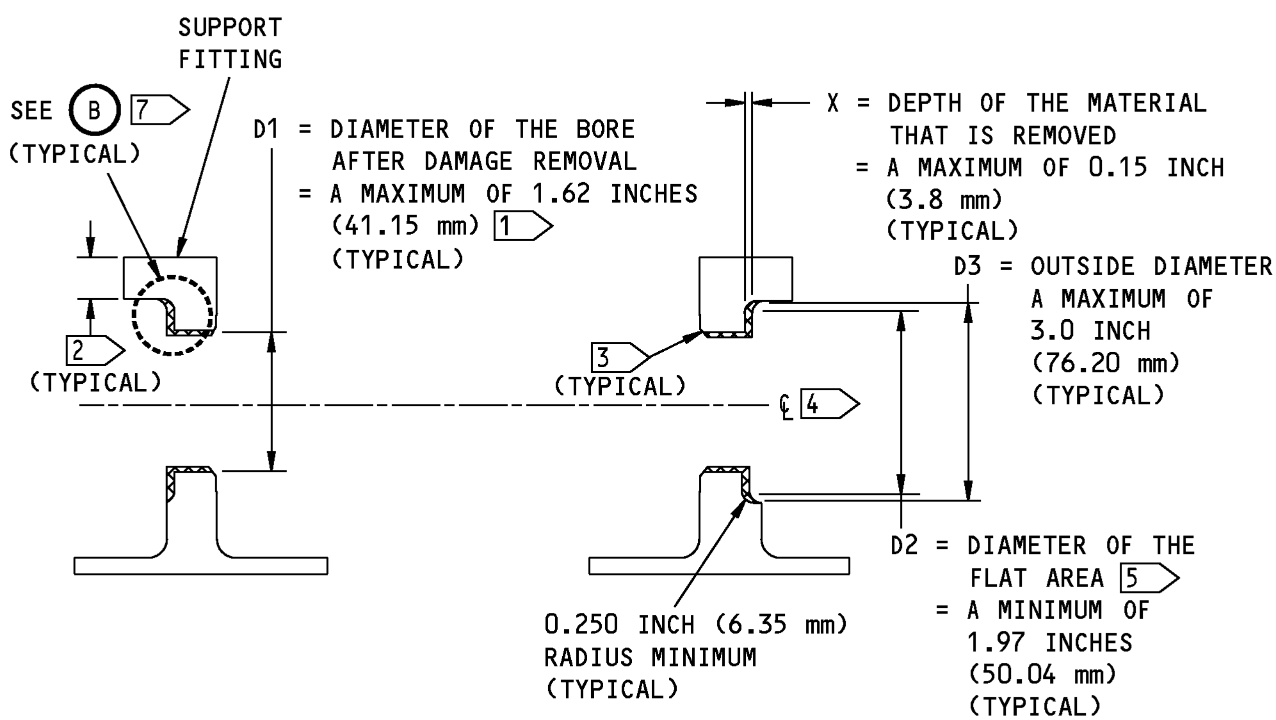
(2) Make the part [3] washer thickness the same as the initial special washer thickness. Refer to Figure 203/REPAIR 2, Detail D and Figure 204/REPAIR 2 to identify the part [3] washer.




- ZONE SHOWS THE TYPICAL LOCATIONS OF THE CORROSION DAMAGES.
THE DAMAGED MATERIAL MUST BE REMOVED AS SHOWN IN SECTION A-A.

REMOVAL OF THE DAMAGE

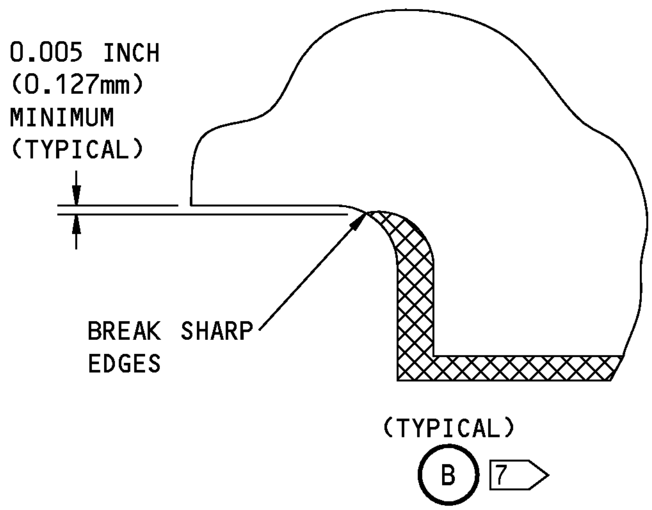
A

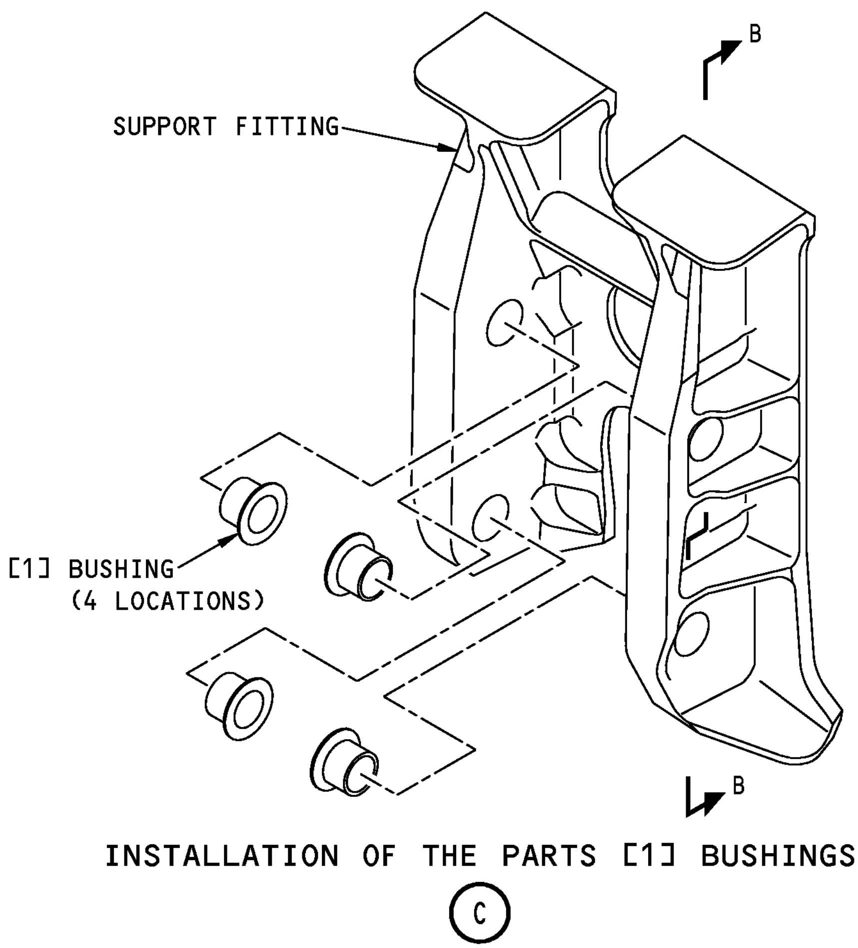



A-A

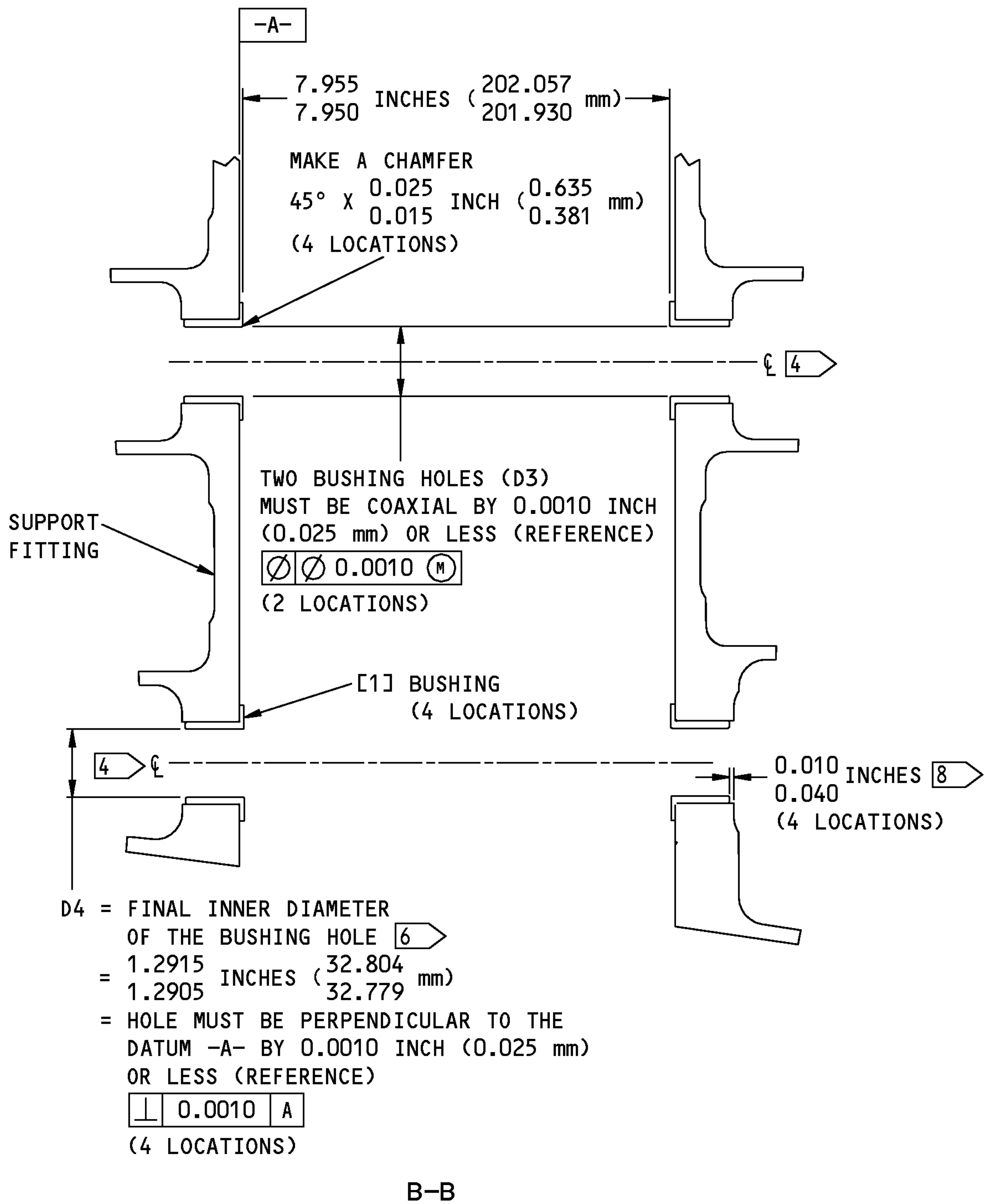
1317270 S0000239552_V1

**Main Landing Gear Forward Trunnion Support Fitting Repair**
**Figure 202 (Sheet 1 of 4)**




**Main Landing Gear Forward Trunnion Support Fitting Repair**
**Figure 202 (Sheet 2 of 4)**




1317272 S0000239555_V1

**Main Landing Gear Forward Trunnion Support Fitting Repair**
**Figure 202 (Sheet 3 of 4)**

NOTES

1 HOLE DIAMETER TOLERANCE MUST NOT BE MORE THAN 0.0008 INCH (0.02 mm).

2 DO NOT REMOVE MATERIAL FROM THE GUSSETS.

3 IF THE BORE WAS REAMED TO AN OVERSIZE DIAMETER, MAKE A CHAMFER AT THIS EDGE. THE CHAMFER IS FROM 0.010 TO 0.020 INCH (0.25 TO 0.50 mm) AND 45 DEGREES.

4 KEEP THE CENTER OF THE HOLES IN THE INITIAL POSITION.

5 MAKE THE INDICATED AREA FLAT AFTER DAMAGE REMOVAL. IT IS NECESSARY FOR CORRECT INSTALLATION OF THE PARTS [2] AND [3] WASHERS.

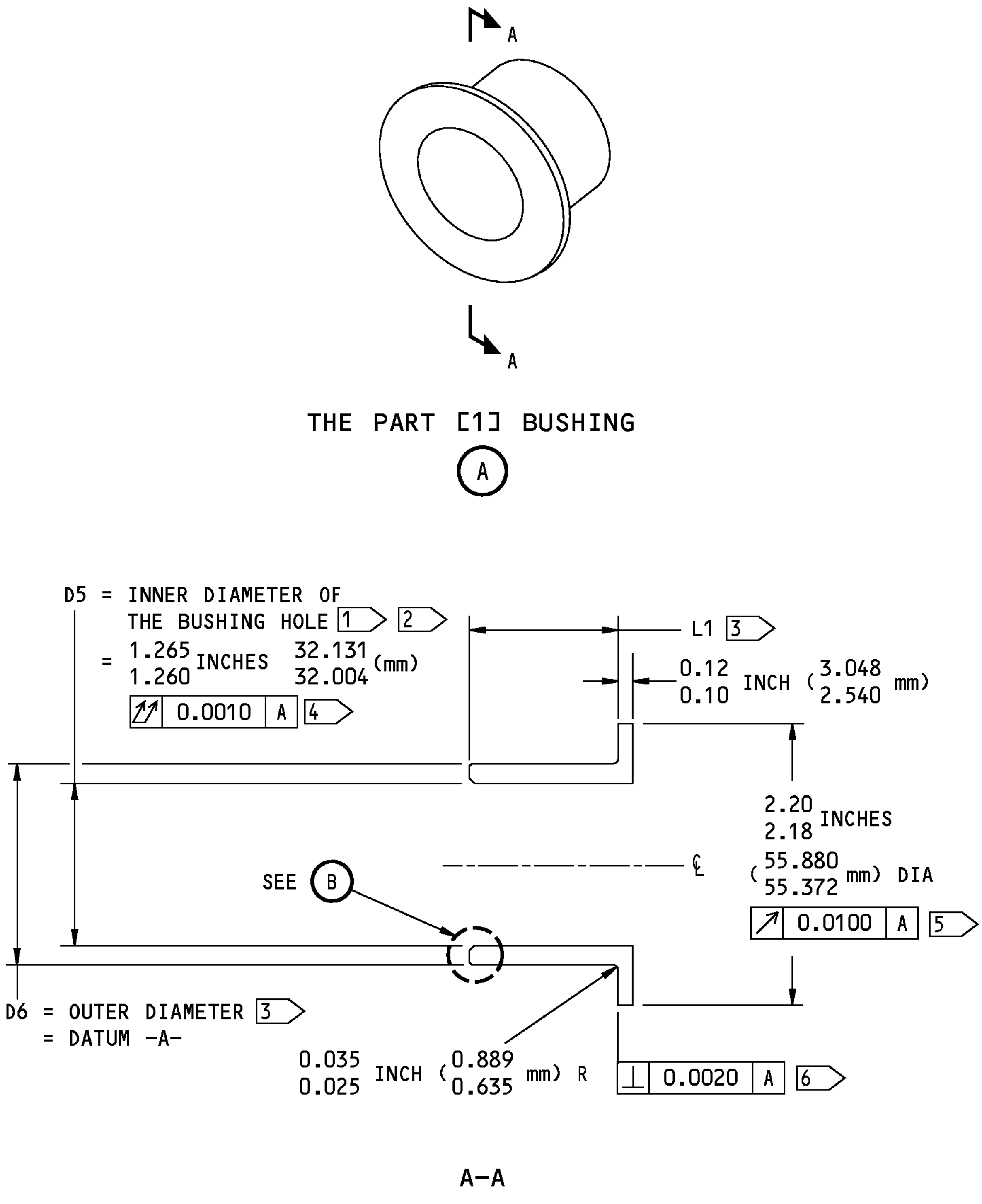
6 REAM AND HONE THE BUSHING HOLE TO THE FINAL INNER DIAMETER AFTER THE INSTALLATION OF THE PART [1] BUSHING.

7 ADJUST THE DIAMETER TO PREVENT DAMAGE TO THE VERTICAL FLANGES.

8 MAKE SURE THAT THE BUSHING IS BELOW FLUSH TO SPECIFIED DIMENSIONS.

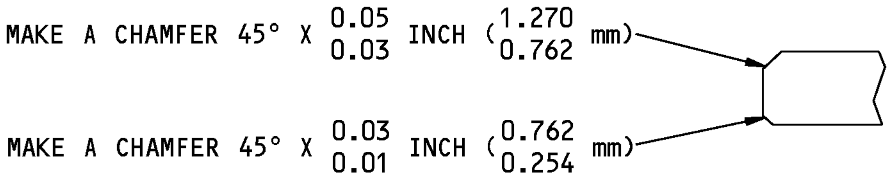
1317273 S0000239556_V1

**Main Landing Gear Forward Trunnion Support Fitting Repair**
**Figure 202 (Sheet 4 of 4)**

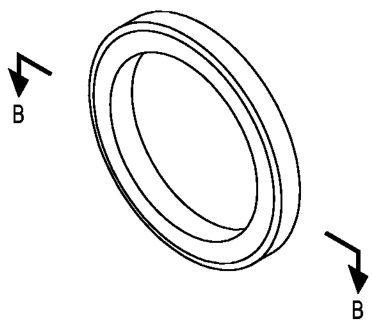



1317274 S0000239557_V1

**Repair Parts**
**Figure 203 (Sheet 1 of 4)**

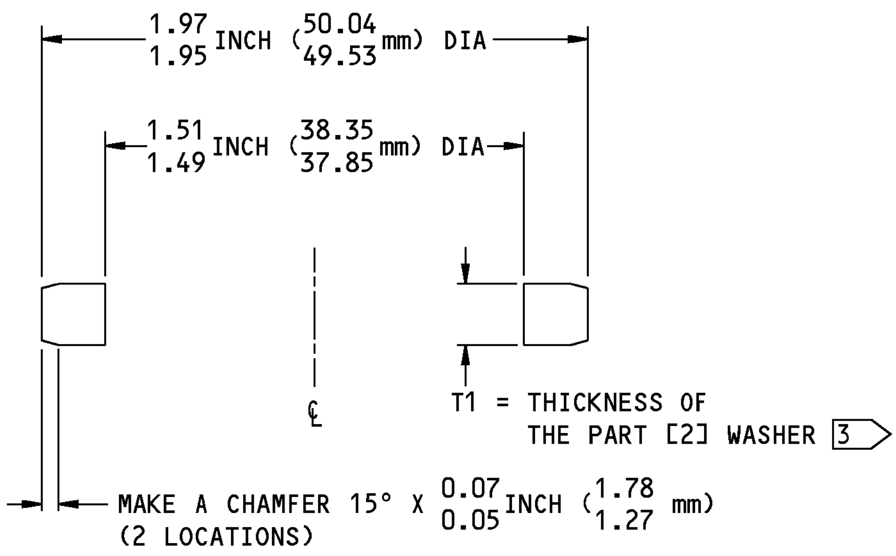



B



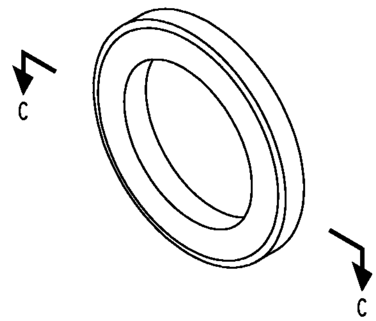

THE PART [2] WASHER

C



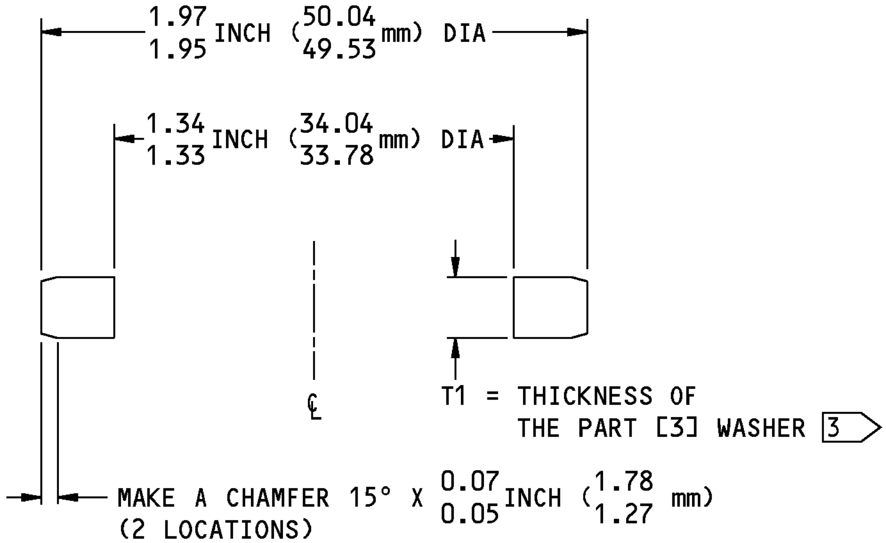

B–B

**Repair Parts**
**Figure 203 (Sheet 2 of 4)**




THE PART [3] WASHER

D




C-C

1562060 S0000289287_V1

Repair Parts
Figure 203 (Sheet 3 of 4)

NOTES

- MAKE THE SURFACE FINISH OF ALL SURFACES (EXEPT AS NOTED) EQUAL TO:
  - 63 MICROINCHES (1.6 MICROMETERS) Ra OR SMOOTHER FOR THE PART [1] BUSHING
  - 125 MICROINCHES (3.2 MICROMETERS) Ra OR SMOOTHER FOR THE PARTS [2] AND 3 WASHER.
- REMOVE ALL SHARP EDGES. MAKE A RADIUS THAT IS 0.005 TO 0.015 INCH (0.127 TO 0.381 mm) AT THE EDGE.
- APPLY CADMIUM PLATE TYPE II, CLASS 2, TO THE PART [1] BUSHING AND THE PARTS [2] AND [3] WASHERS.
- ALL DIMENSIONS ARE SHOWN AFTER THE PLATING PROCEDURE.

1 INNER DIAMETER OF THE BUSHING HOLE BEFORE INSTALLATION IS SHOWN. REAM AND HONE THE BUSHING HOLE TO THE FINAL INNER DIAMETER AFTER THE INSTALLATION OF THE PART [1] BUSHING. REFER TO FIGURE 202, REPAIR 2.

2 IT IS PERMITTED TO MAKE THE SURFACE FINISH OF THE INNER DIAMETER OF THE BUSHING HOLE TO 125 MICROINCHES (3.175 MICROMETERS) Ra OR SMOOTHER. CADMIUM PLATING IS OPTIONAL ON THE INSIDE DIAMETER OF THE BUSHING HOLE.

3 REFER TO PARAGRAPH 5/REPAIR 2 TO CALCULATE THE NECESSARY VALUE.

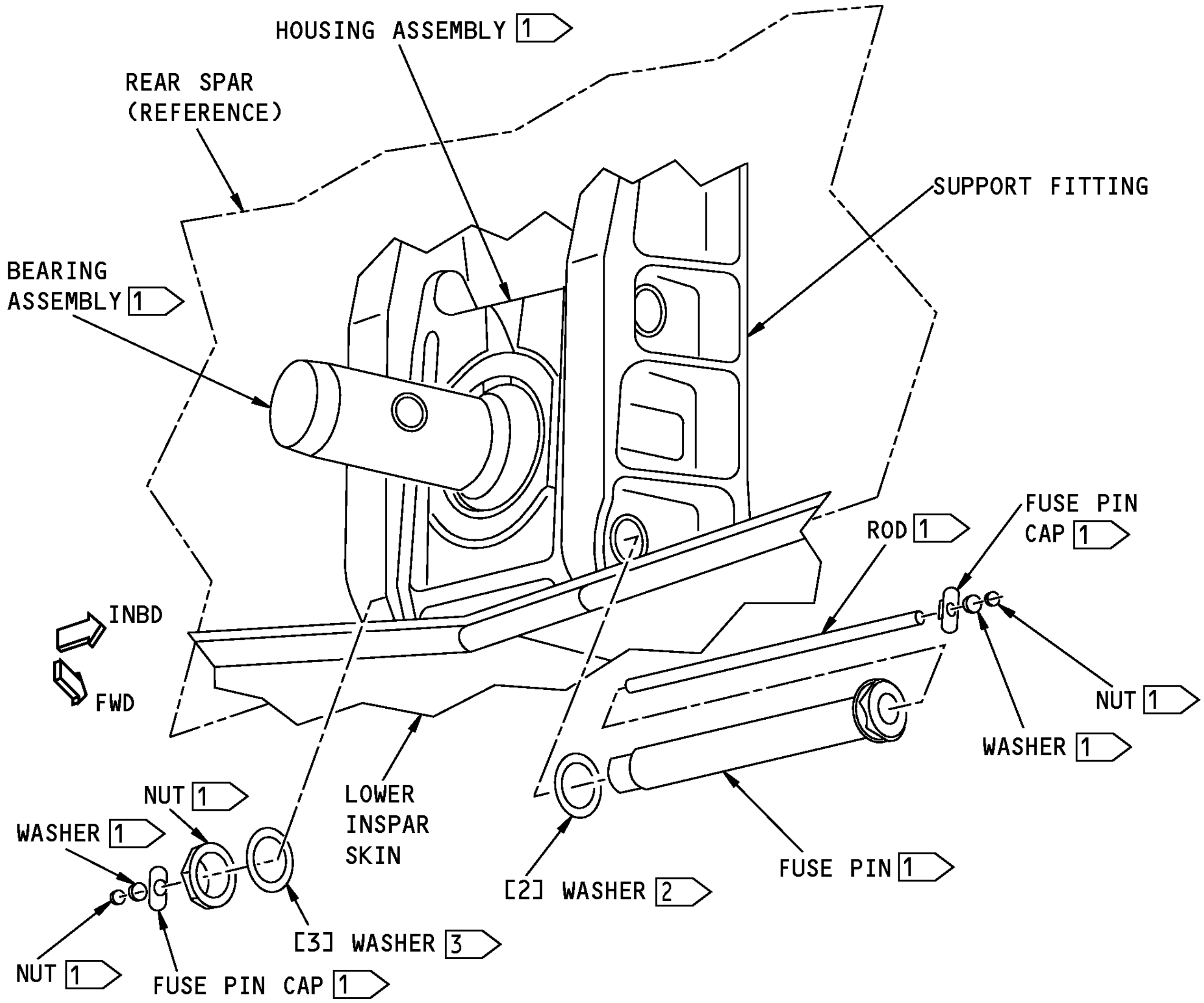
4 TOTAL RUNOUT TOLERANCE TO THE DATUM -A- MUST NOT BE MORE THAN 0.0010 INCH (0.0254 mm).

5 RUNOUT TOLERANCE TO THE DATUM -A- MUST NOT BE MORE THAN 0.0100 INCH (0.254 mm).

6 THE INDICATED SURFACE MUST BE PERPENDICULAR TO THE DATUM -A- WITH A MAXIMUM TOLERANCE OF 0.002 INCH (0.05 mm).

1317276 S0000239559_V1

**Repair Parts**
**Figure 203 (Sheet 4 of 4)**




TYPICAL INSTALLATION OF THE PARTS [2] AND [3] WASHERS

NOTES

1 REFER TO AMM 32-11-83.

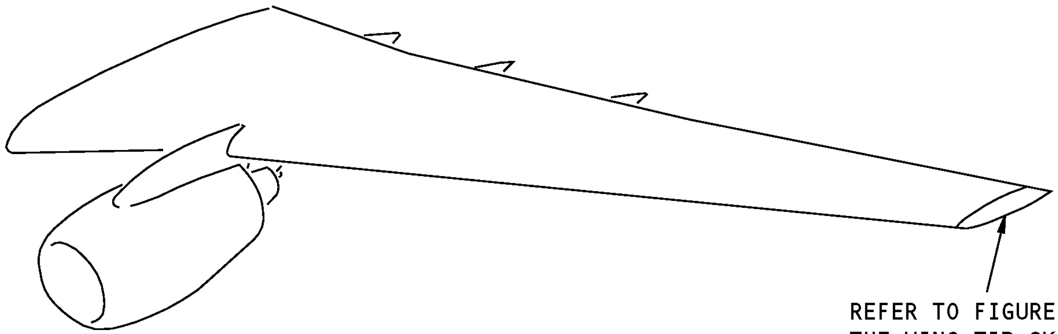
2 REMOVE AND DISCARD THE INITIAL SPECIAL WASHER THEN INSTALL THE PART [2] WASHER AT THE HEAD SIDE OF THE FUSE PIN.

3 REMOVE AND DISCARD THE INITIAL SPECIAL WASHER THEN INSTALL THE PART [3] WASHER AT THE NUT SIDE OF THE FUSE PIN.

1317285 S0000239560_V1

**Installation of the Repair Washer**
**Figure 204**

## IDENTIFICATION 1 - WING TIP SKIN



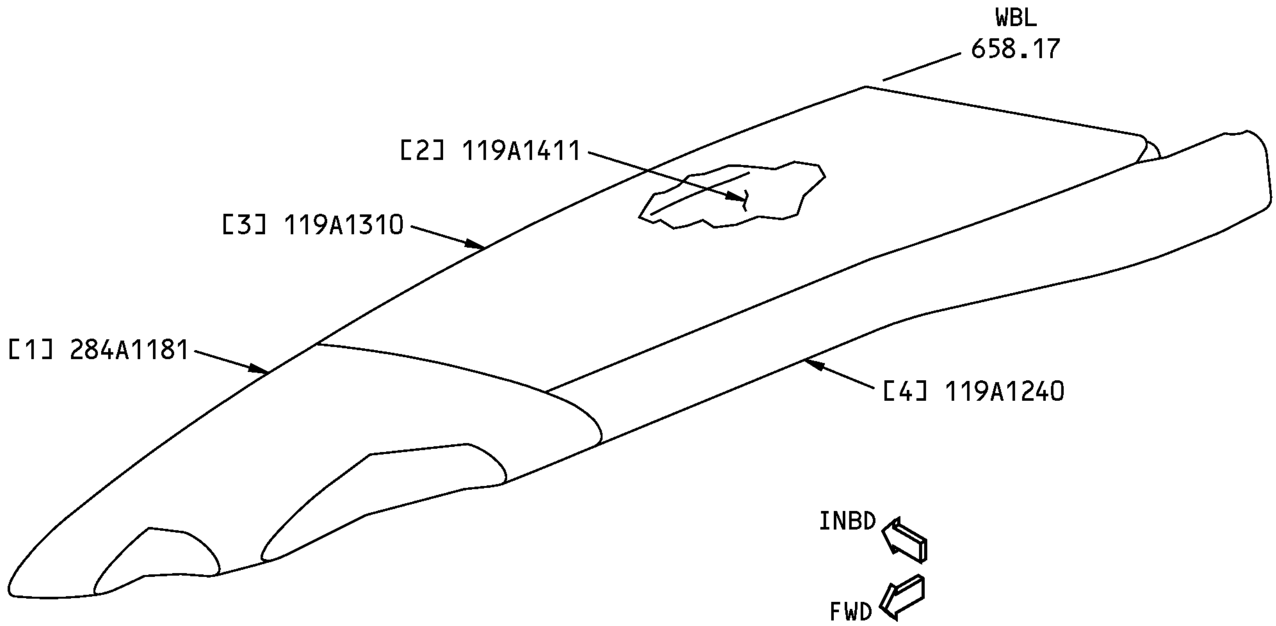

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Tip Skin Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 119A0100 | Wing Tip Installation |
| 119A1200 | Wing Tip Fairing Assembly |
| 119A1500 | Wing Tip Fairing and Lighting Assembly |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

**Wing Tip Skin Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Tip Light Assembly | | Supplied by: Grimes Aerospace Co. 550 Route 55 Urbana, Ohio 43078 | |
| [2] | Wing Tip Lower Skin | 0.063 (1.60) | 7075-T62 bare sheet as given in QQ-A-250/12 | |
| [3] | Wing Tip Upper Skin | 0.063 (1.60) | 7075-T62 bare sheet as given in QQ-A-250/12 | |
| [4] | Torque Tube | | A357.0-T6 as given in MIL-A-21180, Class 10 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - WINGLET SKIN

1. **Applicability**

   A. Identification 2 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

   **NOTE**: Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

   (1) For airplanes with cum line numbers 1 thru 1302:

      (a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

   (2) For airplanes with cum line numbers 1303 and on:

      (a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

   B. Refer to Table 1 for the reference drawings and Table 2 for the List of Materials.

**Table 1**:

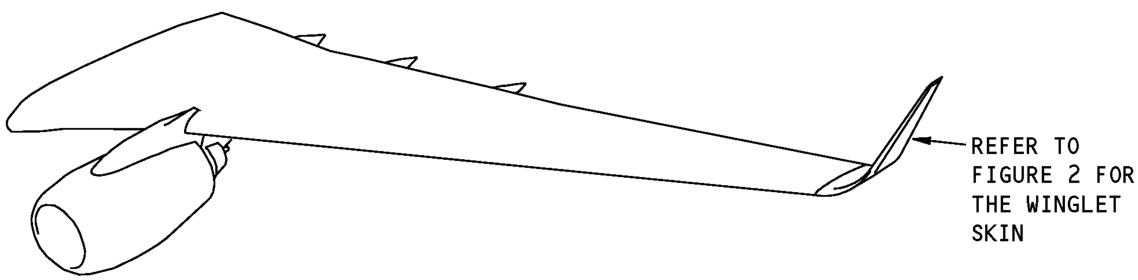
| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 737-0002 | Winglet Installation |
| 737-0010 | Winglet Final Assembly |
| 737-0310 | Winglet Upper Skin Installation |
| 737-0330 | Winglet Lower Skin Installation |

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Leading Edge Skin - Inboard | 0.063 (1.60) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [2] | Leading Edge Skin - Mid | 0.063 (1.60) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [3] | Leading Edge Skin - Outboard | 0.063 (1.60) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [4] | Bonded Upper Skin Panel | | Fiberglass/Graphite/Epoxy Honeycomb Sandwich<br>Fiberglass/epoxy Fabric as given in BMS 8-139, Style 120, Class 1<br>Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190<br>Graphite/Epoxy Fabric as given in 8-256, Type IV, Class 2, Style 3K-70-PW | |
| | Skin | | Refer to APB drawing 737-0311 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class 4, Type V and VI, Grade 3.0 | |
| [5] | Bonded Lower Skin Panel | | Fiberglass/Graphite/Epoxy Honeycomb Sandwich<br>Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1<br>Graphite/epoxy tape as given in BMS 8-256, Type II, Class 1, grade 190 | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | | | Graphite/Epoxy Fabric as given in 8-256, Type IV, Class 2, Style 3K-70-PW | |
| | Skin | | Refer to APB drawing 737-0331 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class 4, Type V and VI, Grade 3.0 | |
| [6] | Tip Fairing | 0.063 (1.60) | 6061-T62 sheet as given in QQ-A-250/11 | |
| [7] | Inboard Trailing Edge Wedge | | Epoxy sheet molding as given in BMS 8-327 A, Type 1 | |
| [8] | Outboard Trailing Edge Wedge | | 2024-T3511 extrusion as given in QQ-A-200/3 | |

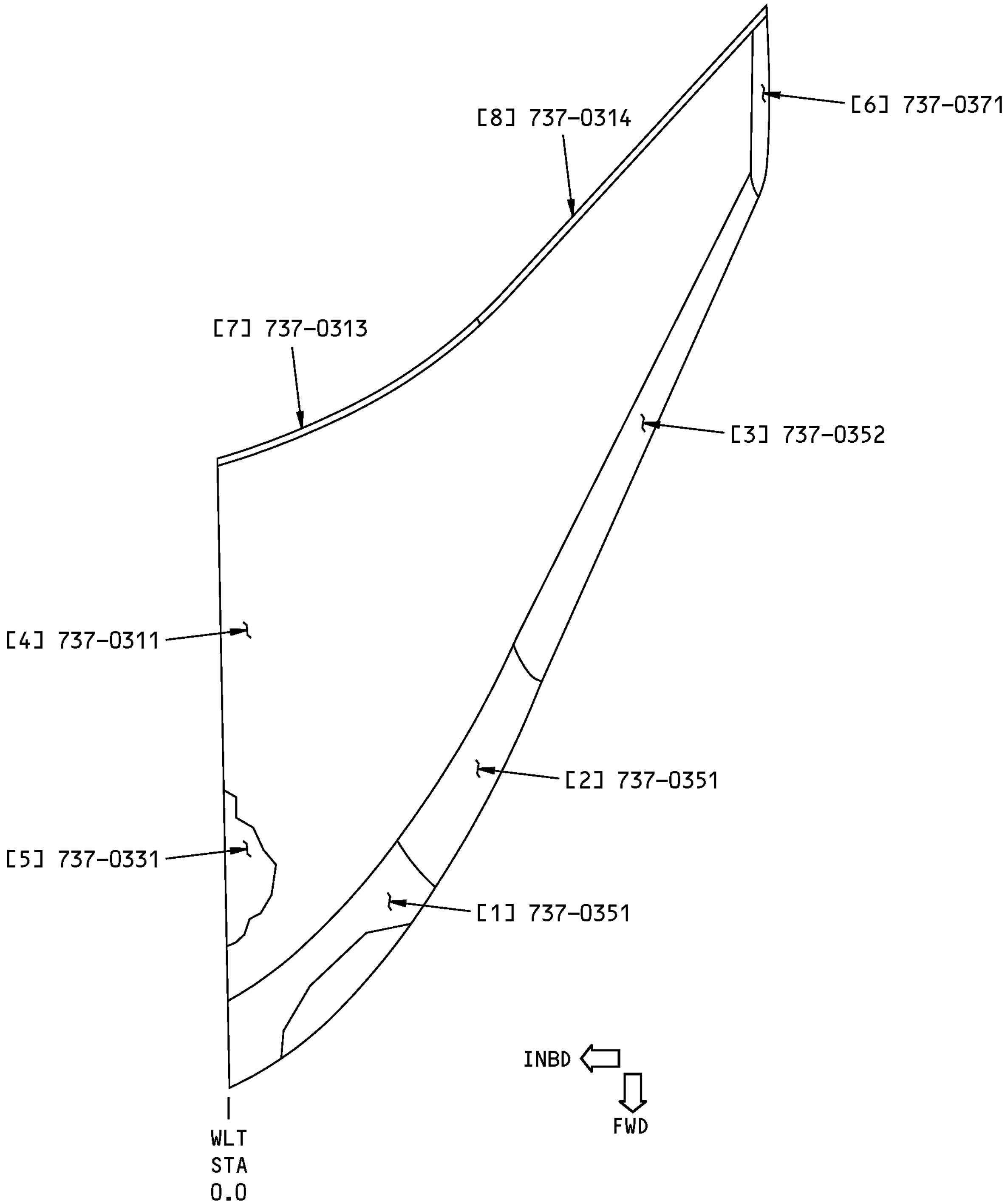
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Winglet Skin Location**
**Figure 1**




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Winglet Skin Identification**
**Figure 2**

## ALLOWABLE DAMAGE 1 - WING TIP SKIN

1. **Applicability**

   A. This subject gives the allowable damage limits for the wing tip skins shown in Wing Tip Skin Location, Figure 101/ALLOWABLE DAMAGE 1.

2. **General**

   A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

   B. Remove the damage as necessary.

      (1) Refer to 51-10-02 for the investigation and removal of damage.

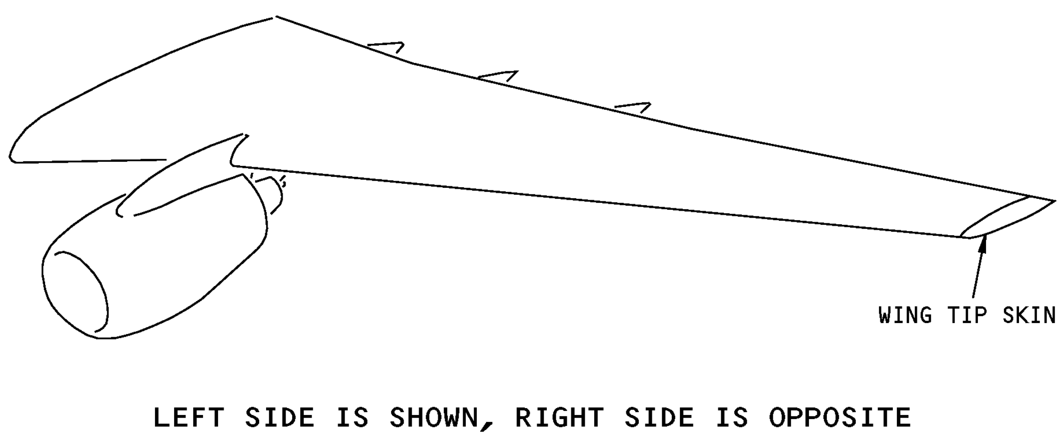
      (2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

      (3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

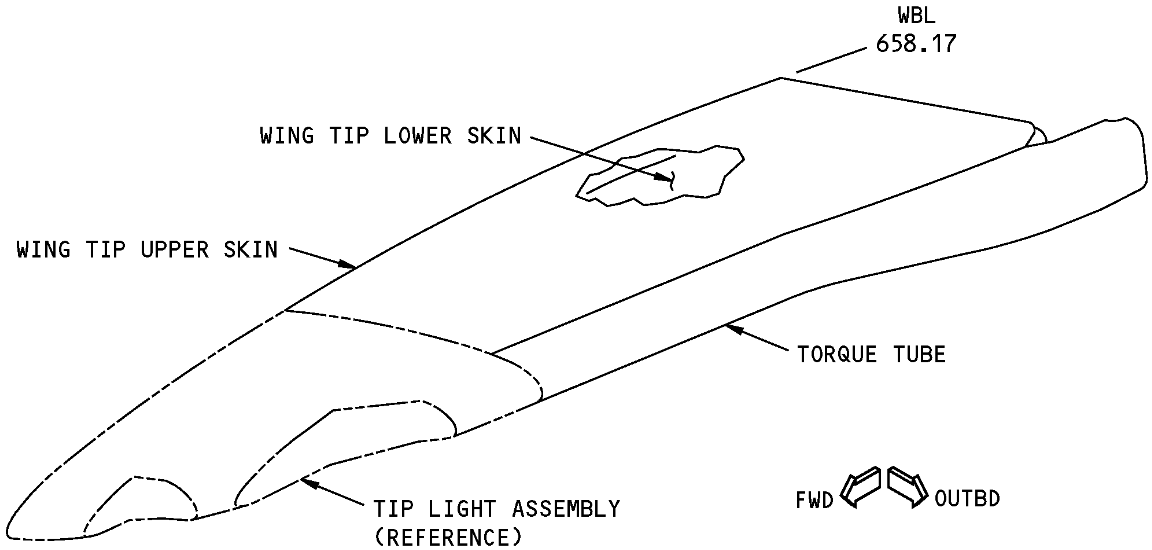
   C. After you remove the damage, do the steps that follow:

      (1) Apply a chemical conversion coating to reworked areas. Refer to 51-20-01.

      (2) Apply one layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.




**Wing Tip Skin Location**
**Figure 101**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: ALL THE PARTS ARE MADE OF ALUMINUM.

**Wing Tip Skin**
**Figure 102**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage limits for the Wing Tip Skin

A. Cracks:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

B. Nicks, Gouges, Scratches, and Corrosion:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

C. Dents:

(1) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F .

(2) If the damage depth is more than the 0.125 inch and less than 0.30 inch, do the steps that follow:

(a) Fill or rework the damage as given in 51-70-01.

(b) Apply aluminum foil tape (speed) tape.

(c) Record the damage location.

(d) Do an inspection of the damage location at normal maintenance intervals.

(e) Repair the damage before 24 months have occurred.

D. Holes and Punctures:

(1) The damage is permitted if it is:

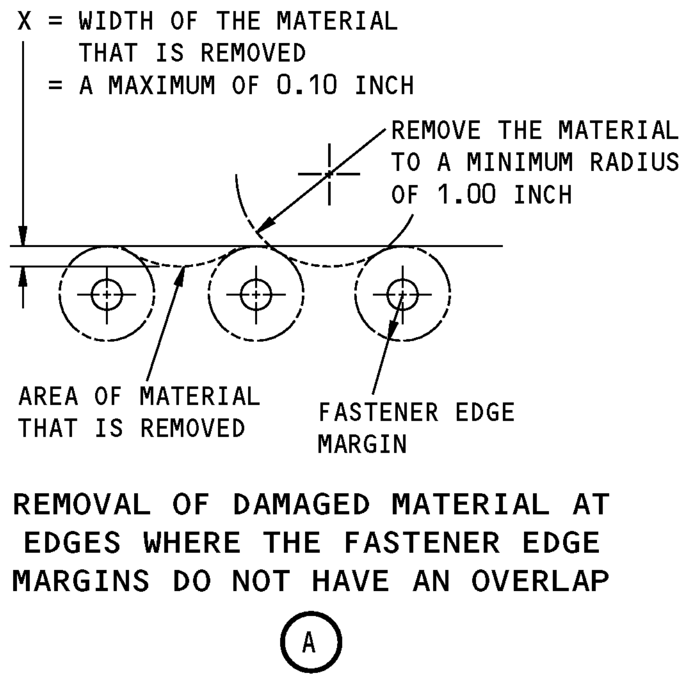
(a) A maximum of 0.25 inch in diameter.

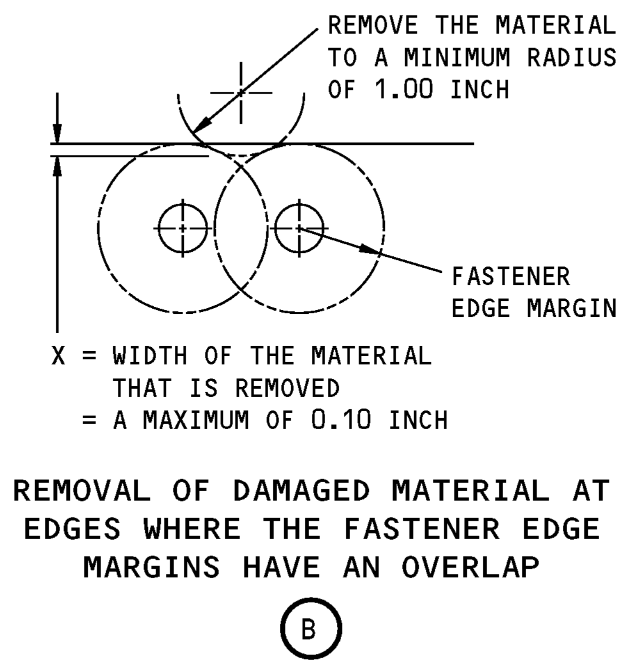
(b) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

(c) Filled with a rivet without sealant.

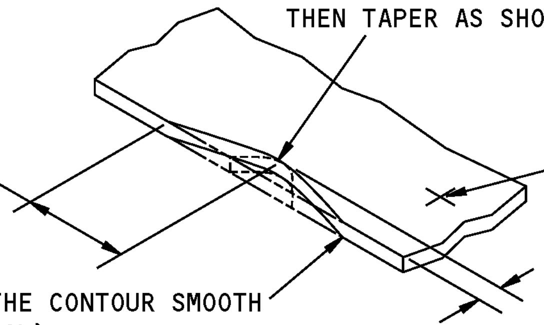
1) For the wing tip skins, use a 2117-T3 or 2117-T4 protruding head rivet.

2) For the torque tube, use a 5056-H32 countersink head rivet.



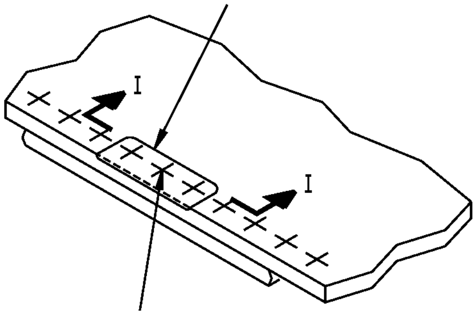

**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 3)**




X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

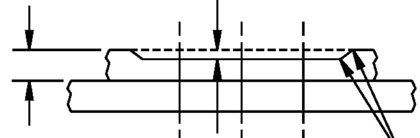
REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C



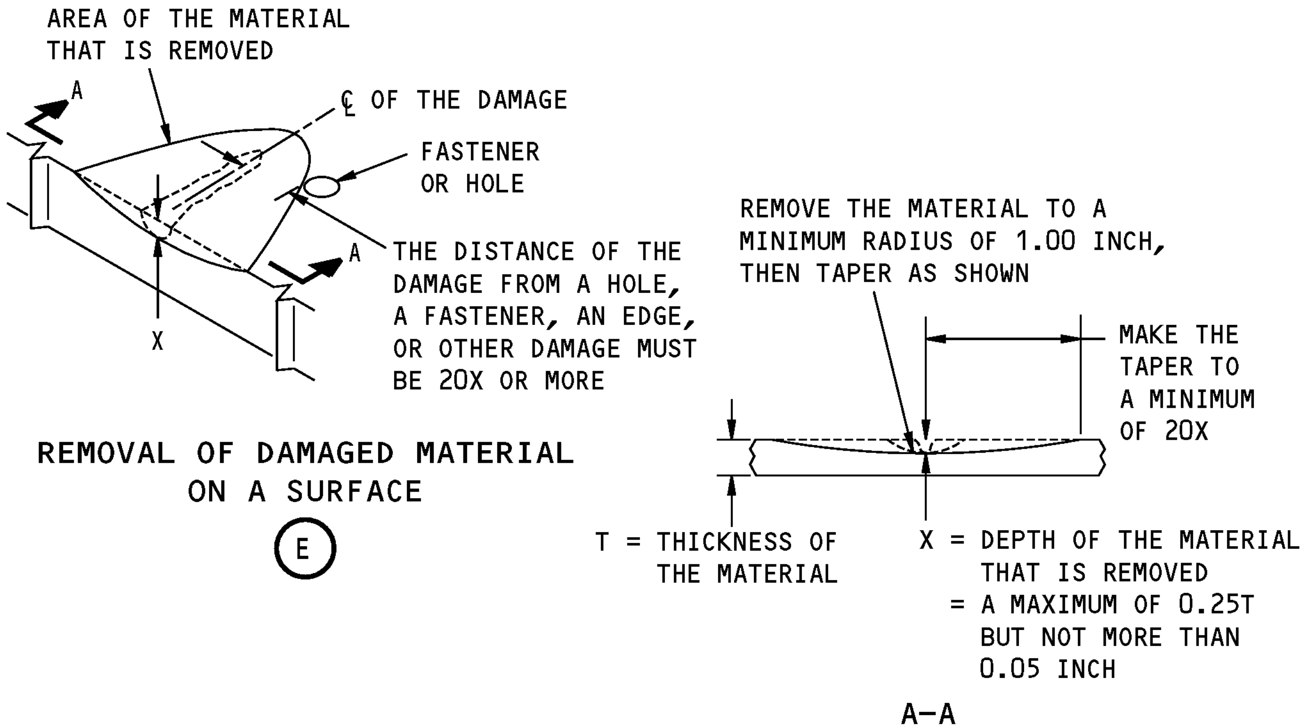

REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

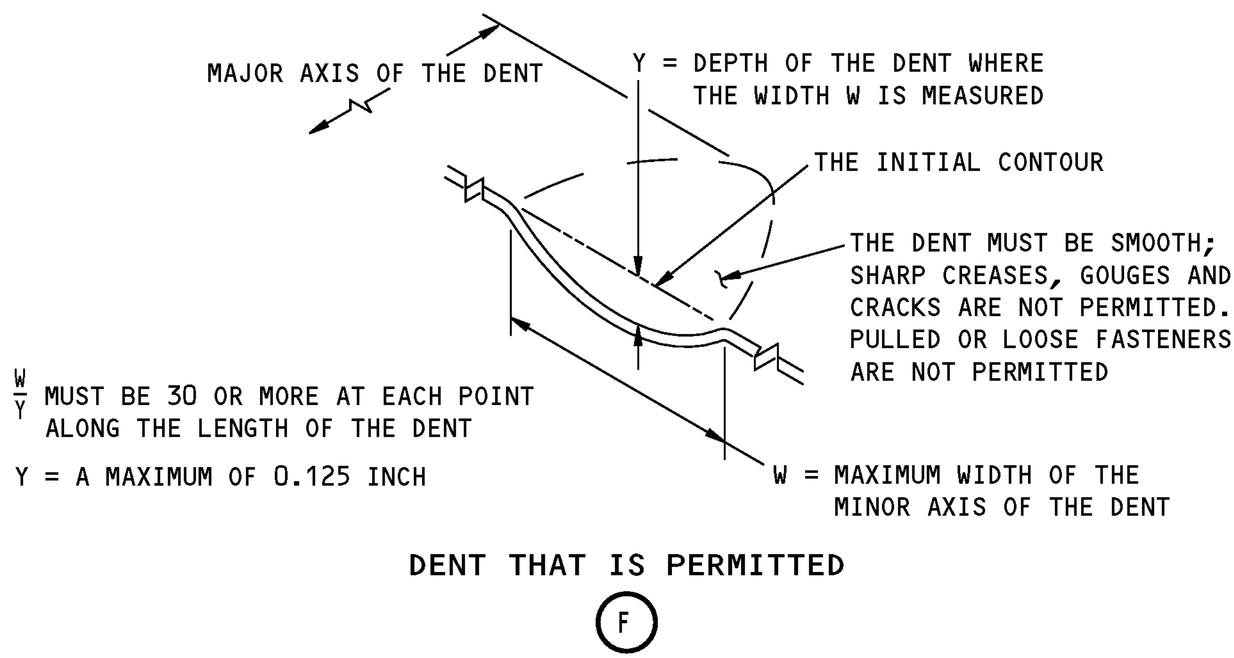
D




I-I

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 3)**




**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 3)**

## ALLOWABLE DAMAGE 2 - WINGLET SKIN

1. **Applicability**

A. Allowable Damage 2 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Allowable Damage 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

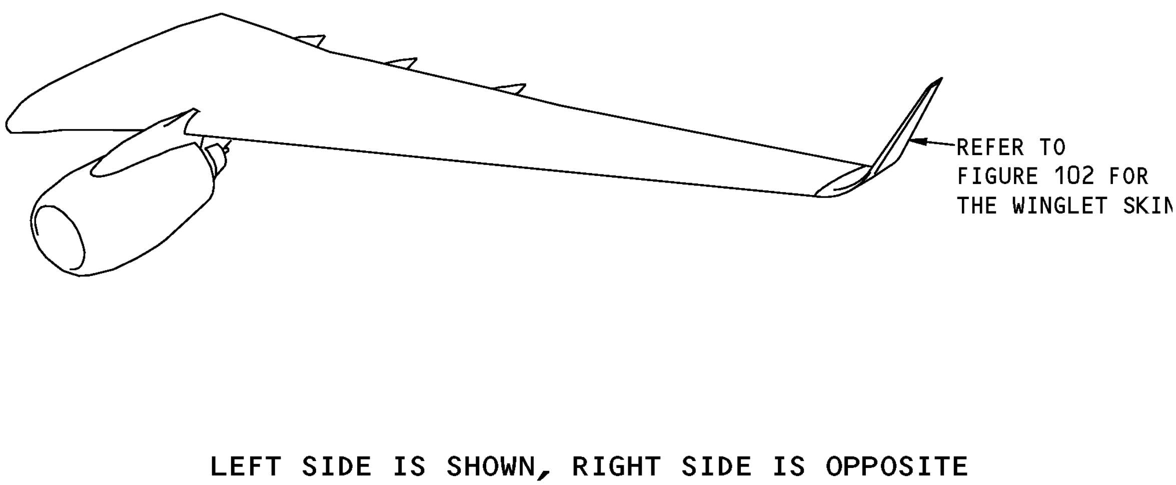
(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

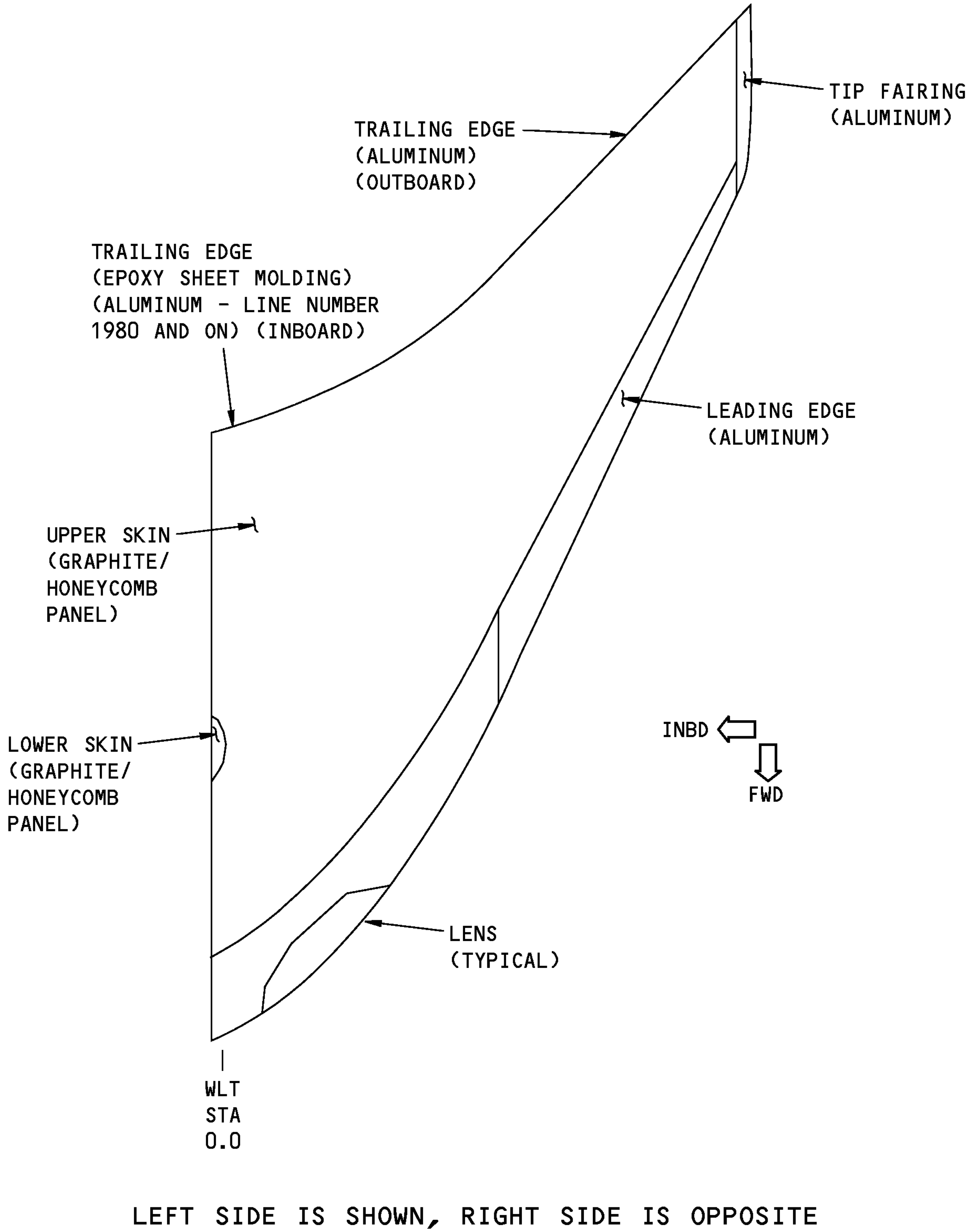
(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. This subject gives the allowable damage limits for the winglet skins shown in Winglet Skin Location, Figure 101/ALLOWABLE DAMAGE 2.




**Winglet Skin Location**
**Figure 101**




**Winglet Skin Allowable Damage**
**Figure 102**

**2. General**

A. For damage to all parts other than the winglet upper and lower skins do as follows:

(1) Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

(2) Remove the damage as necessary.

(a) Refer to 51-10-02 for the investigation and removal of damage.

(b) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(3) After you remove the damage, do the steps that follow. The steps that follow do not apply to the inboard trailing edge wedge.

**NOTE**: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

(a) Apply a chemical conversion coating to reworked areas. Refer to 51-20-01.

(b) Apply one layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in economic performance of the airplane. Refer to AERODYNAMIC SMOOTHNESS, 51-10-01.

B. For damage to the winglet upper and lower skins do as follows:

(1) Refer to Paragraph 5./ALLOWABLE DAMAGE 2 for the allowable damage limits.

(2) Do an inspection of the damaged areas to find the length, width, and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

(a) Remove all contaminates and water from the structure. Use a vacuum and heat to a maximum temperature of 125 degrees F (52 degrees C).

(b) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C for the definitions of the length, width, and depth of damage.

(c) Refer to Definitions of the Facesheets, Figure 104/ALLOWABLE DAMAGE 2 for the definitions of the facesheets of a honeycomb core.

(3) Seal all permitted damage that are not more than one ply deep. Seal the damage with one of the two procedures that follow:

(a) Make a temporary seal.

1) Apply aluminum foil tape (speed tape)

2) Keep a record of the location

3) Make sure the tape is in satisfactory condition at normal maintenance intervals.

4) Replace the tape if more damage occurs.

(b) Make a permanent seal.

1) Apply BMS 8-201, BMS 8-207, or BMS 8-301 epoxy resin to the area.

2) Apply one layer of BMS 10-79, Type III primer. Refer to SOPM 20-44-04.

3) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

**NOTE**: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

(4) Seal all of the damage areas that are more than one ply in depth. Refer to the allowable damage limits as given in Paragraph 4. Seal the damage as follows:

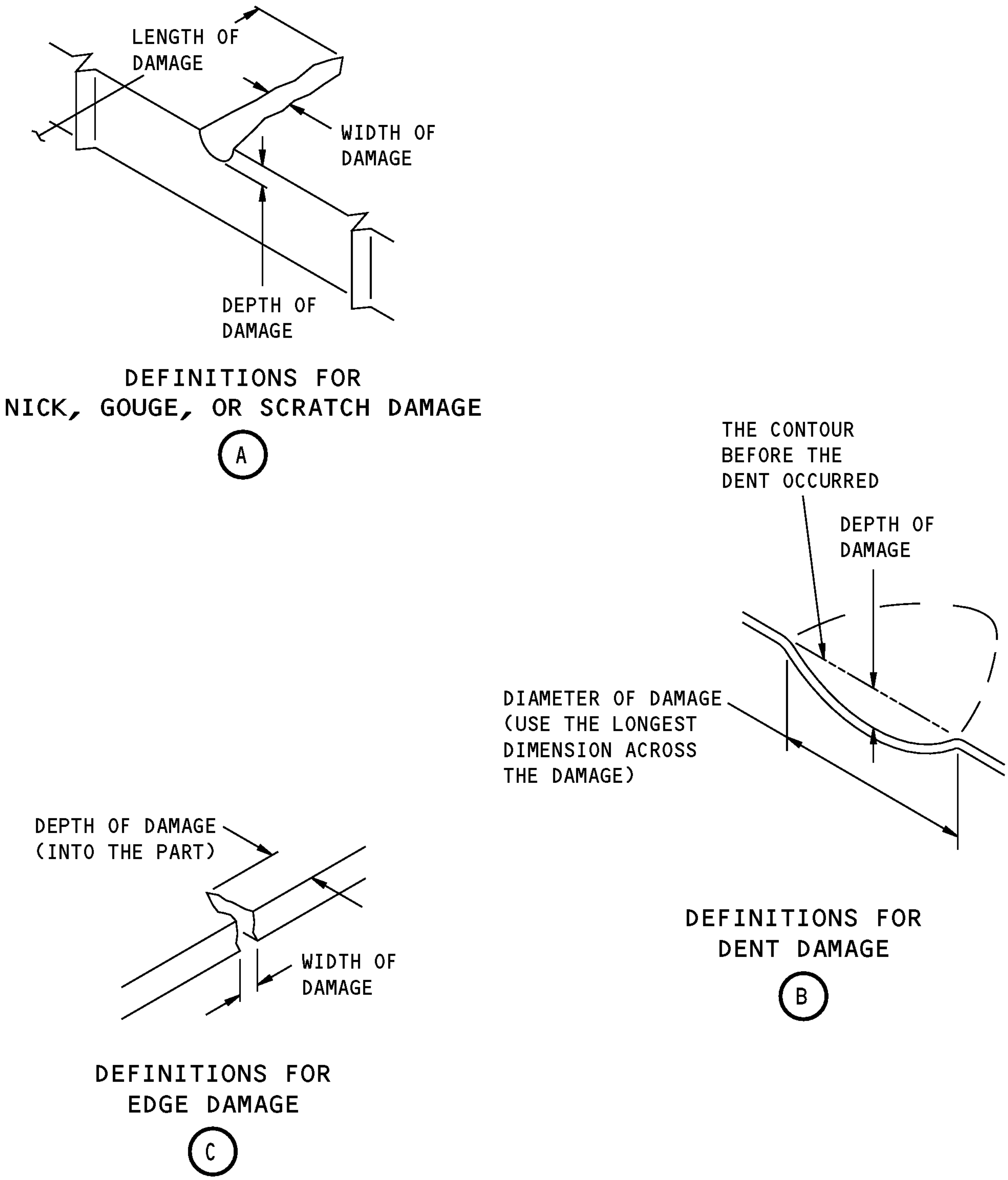
(a) Use a vacuum and heat to remove the moisture from the solid laminate or honeycomb cells. Refer to REPAIR PROCEDURES FOR WET LAYUP MATERIALS, PAGEBLOCK 51-70-04, REPAIR.

(b) Make a temporary seal with aluminum foil tape (speed tape).

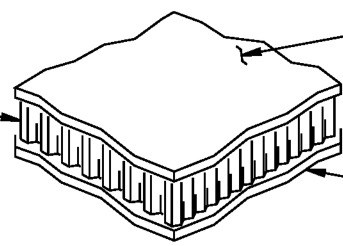
(c) Keep a record of the location.

(d) Do a permanent repair in 24 months or less.

(5) Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in economic performance of the airplane. Refer to 51-10-01.




**Damage Definitions**
**Figure 103**




**Definitions of the Facesheets**
**Figure 104**

## 3. References

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-70-04, REPAIR P/B REPAIR | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

## 4. Allowable Damage limits for the Winglet Aluminum and Epoxy Sheet Molding Parts

A. Cracks:

(1) Edge cracks are permitted as given in Figure 105, Details A, B, and C.

B. Nicks, Gouges, Scratches, and Corrosion:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 2, Detail D .

C. Dents:

(1) The damage is permitted as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 2, Detail F .

D. Holes and Punctures for the Leading Edge Skin and Tip Fairing:

(1) The damage is permitted if it is:

(a) A maximum of 0.25 inch in diameter.

(b) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

(c) Filled with an aluminum rivet installed wet with BMS 5-95 sealant. Refer to 51-20-05.

1) Use a 2117-T3 or 2117-T4 protruding head rivet.

E. Holes and Punctures for the Trailing Edge Wedge is not permitted.

## 5. Allowable Damage limits for the Winglet Graphite Upper and Lower Skins

A. Cracks:

(1) Edge cracks are permitted as given in Figure 105, Details A, B, and C.

B. Nicks, Gouges, and Scratches:

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers but not the carbon fibers are permitted.

C. Dents are not permitted.

D. Holes and Punctures are permitted if the damage is:

(1) A maximum of 1 damage location for each winglet installation

(2) A maximum diameter of 2.0 inches

(3) A minimum of 6.0 inches from a part edge, fastener hole, or other damage

(4) Sealed and repaired as given in Paragraph 2.

E. Delamination is permitted if the damage is:

(1) A maximum of 1 damage location for each winglet installation

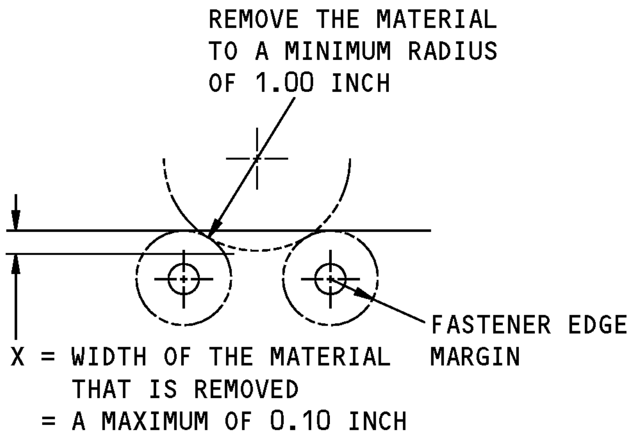
(2) A maximum diameter of 2.0 inches

(3) A minimum of 6.0 inches from a part edge, fastener hole, or other damage

(4) Sealed and repaired as given in Paragraph 2.

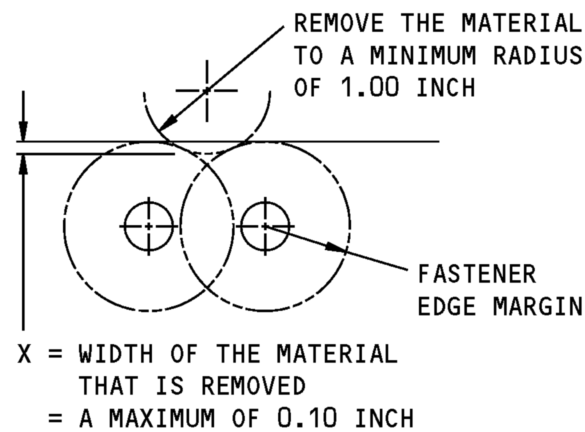
**6. Allowable Damage limits for the Winglet Lens**

A. Damage is not permitted.



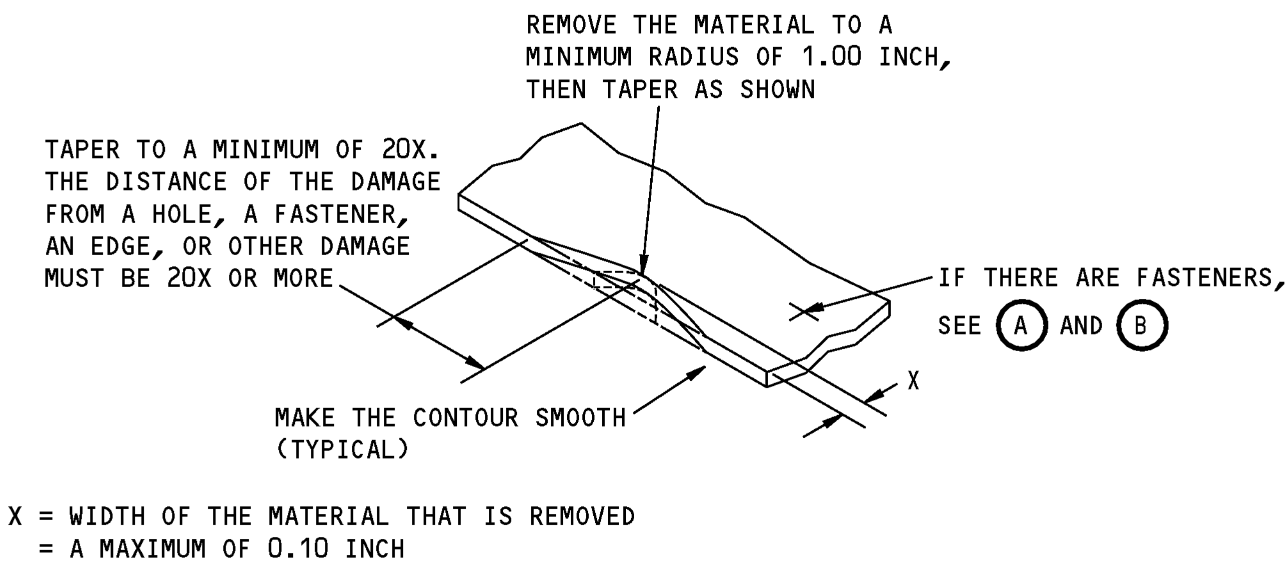

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

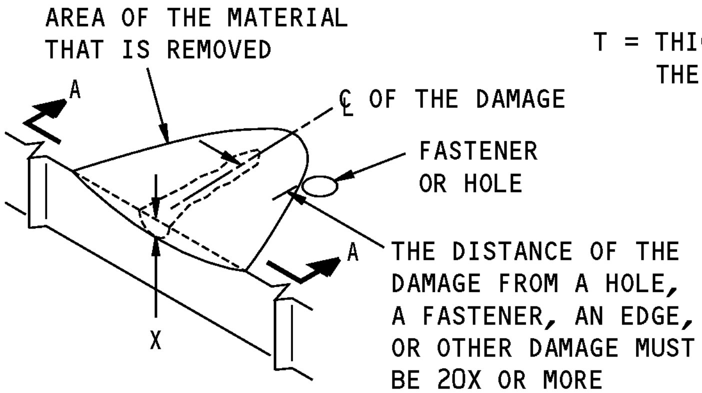
B

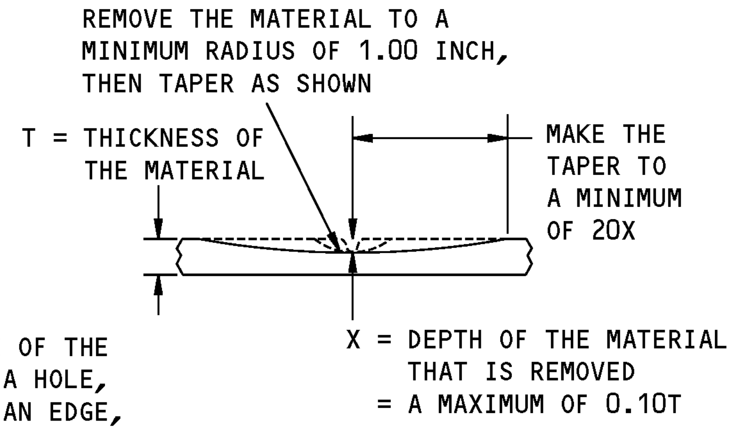



REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

**Allowable Damage Limits**
**Figure 105 (Sheet 1 of 3)**

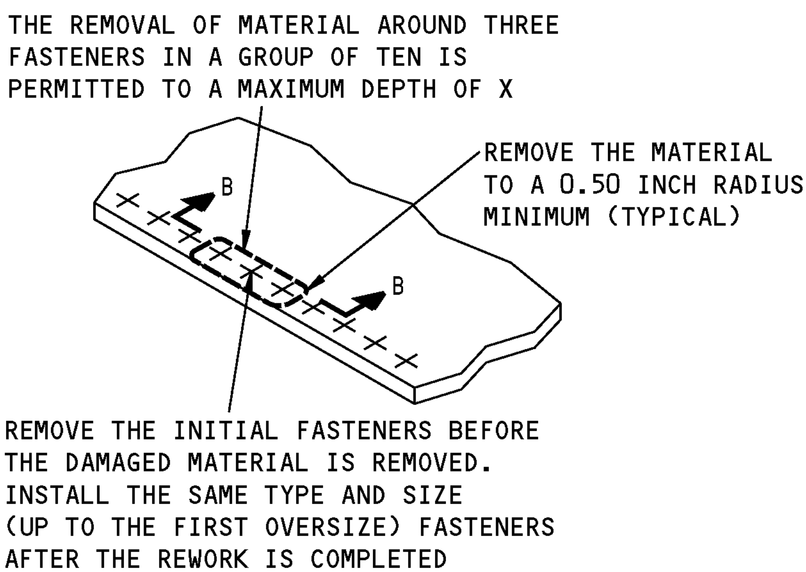



A-A

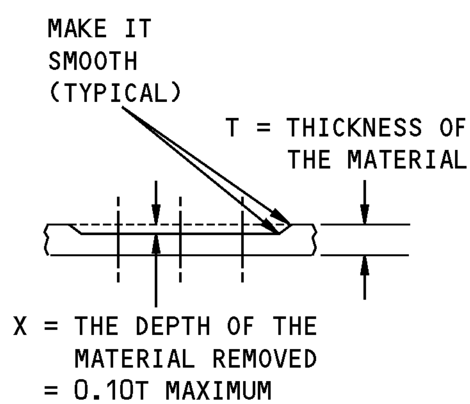
REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D



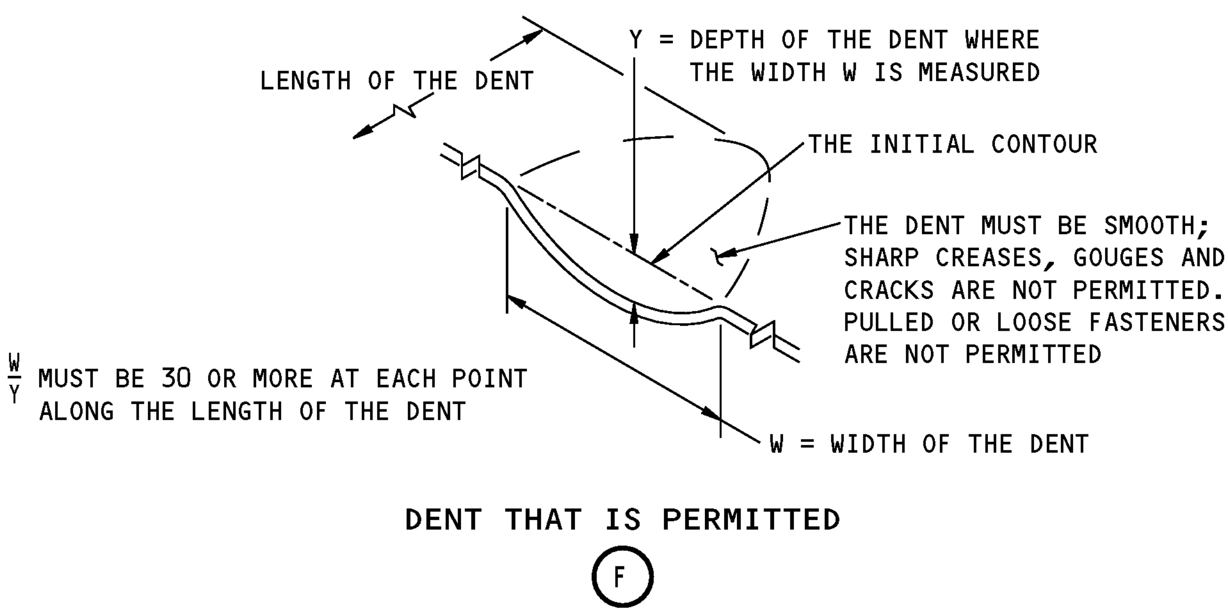

REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

E




B-B

**Allowable Damage Limits**
**Figure 105 (Sheet 2 of 3)**




**Allowable Damage Limits**
**Figure 105 (Sheet 3 of 3)**

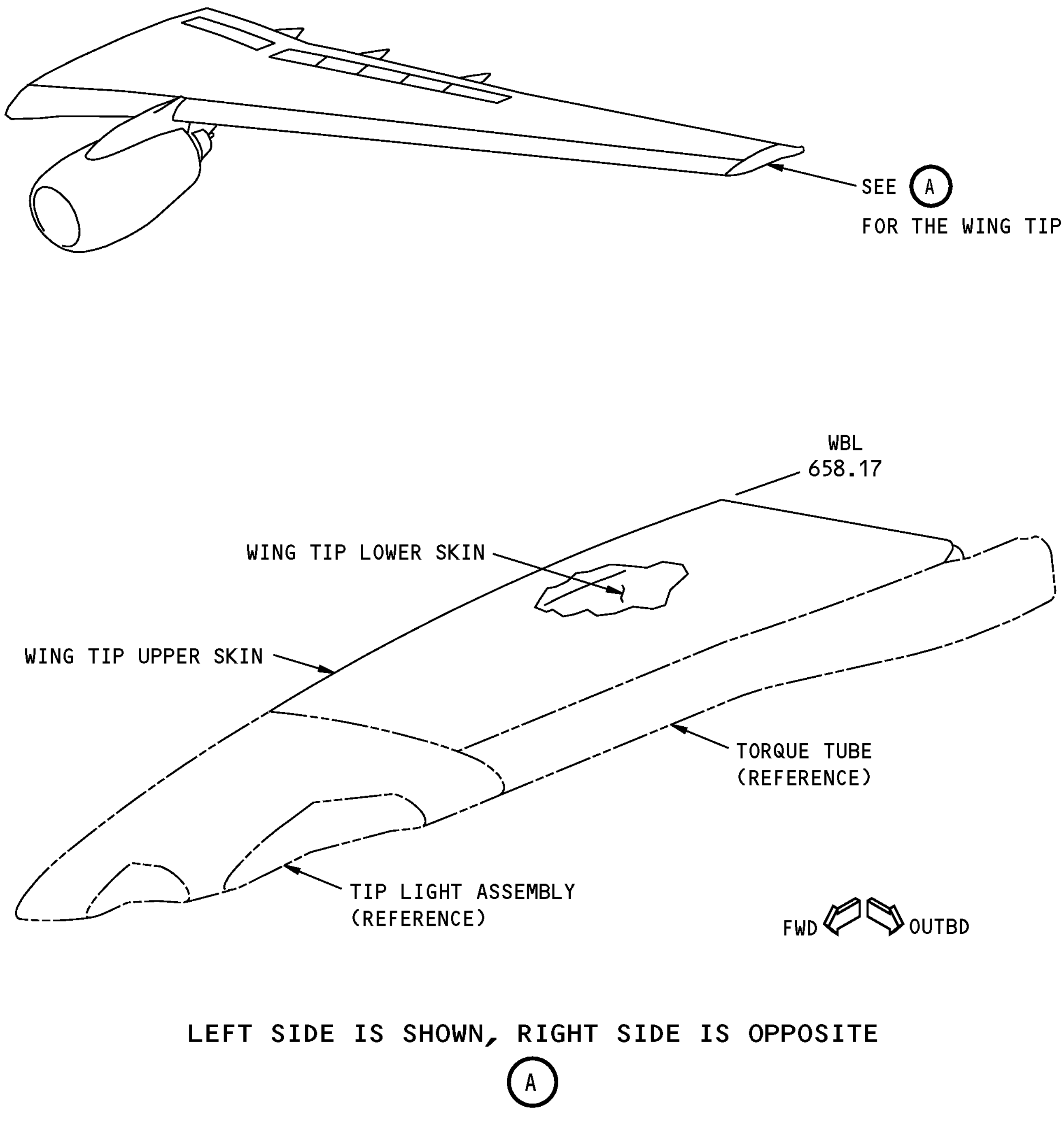
## REPAIR 1 - EXTERNAL REPAIR OF THE WING TIP SKIN

1. **Applicability**

   A. Repair 1 is applicable to damage to the upper and lower skins of the wing tip as shown in Wing Tip Skin Location, Figure 201/REPAIR 1.

2. **General**

   A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.




**Wing Tip Skin Location**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| AMM 57-21-11/401 | Wing Tip Removal/Installation |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

### 4. Repair Instructions

A. Remove the wing tip if necessary. Refer to AMM 57-21-11/401.

B. Remove the damage.

(1) If there is a crack, do one of the steps that follow:

(a) Drill a stop hole at the ends of a crack.

1) Drill a 0.25 inch diameter stop hole at the ends of the crack that do not end at a fastener hole.

2) Make sure you have removed the ends of the crack. Refer to 51-10-02.

(b) Cut and remove the damage. Refer to 51-10-02.

1) Refer to Wing Tip Skin - External Repair, Figure 202/REPAIR 1 for the shape of the cut out.

2) Make the corner radii of the cut a minimum of 0.50 inch.

3) Make sure there is a minimum of two rows of repair fasteners around the edges of the cut out.

C. Make the part [1] doubler as shown in Wing Tip Skin - External Repair, Figure 202/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair material.

(1) Make the contour of the part [1] doubler the same as the initial contour of the skin.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Doubler | 1 | Use bare or clad 7075-T6 that is 0.071 inch thick |

D. Assemble the part [1] doubler and drill the fastener holes.

(1) Do not countersink the fastener holes more than 80 percent of the initial skin thickness.

(a) This will prevent a knife-edge condition of the part [1] doubler.

E. Remove the part [1] doubler.

F. Remove all nicks, scratches, burrs, and sharp edges from the part [1] doubler.

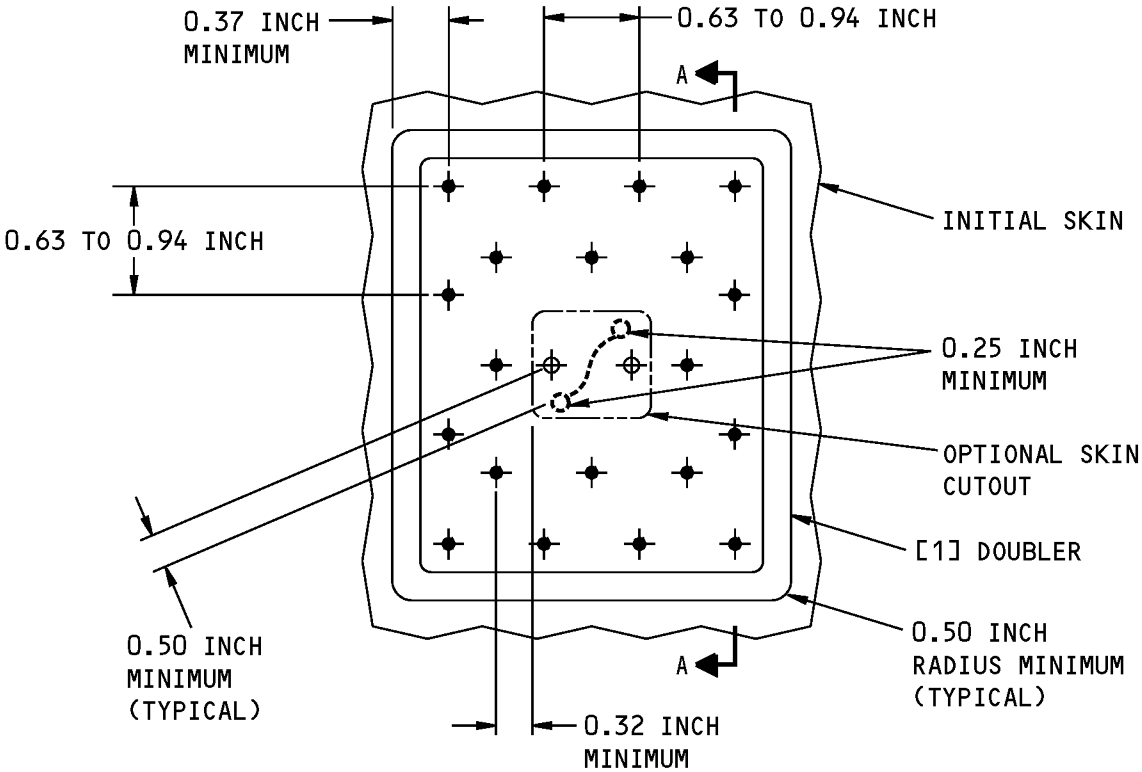
G. Apply a chemical conversion coating to the part [1] doubler and to the bare surfaces of the skin.

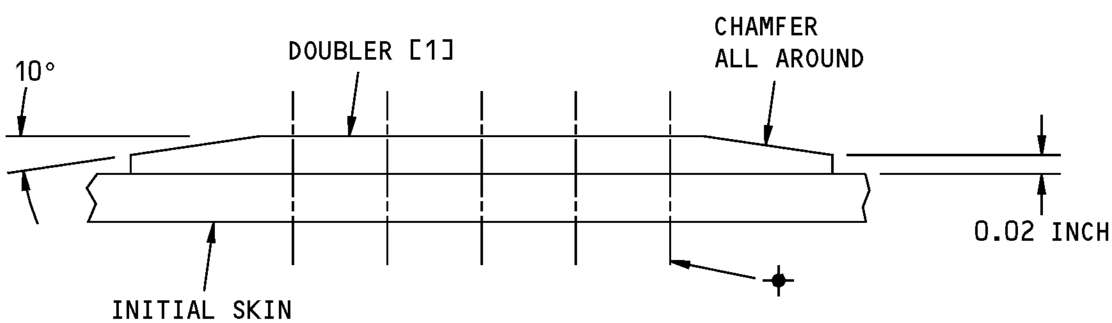
H. Apply one layer of BMS 10-79, Type II primer to the part [1] doubler and to the bare edges of the initial skin. Refer to 51-20-01.

I. Install the part [1] doubler with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

J. Install the fasteners wet with BMS 5-95 sealant.

K. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.




A-A

FASTENER SYMBOLS

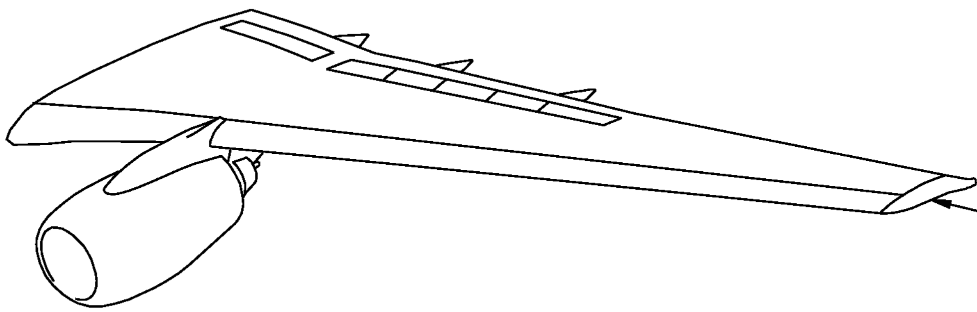
- REPAIR FASTENER LOCATION. INSTALL A BACR15FP5E BLIND RIVET.
- REPAIR FASTENER LOCATION. INSTALL A BACR15FP5E BLIND RIVET. THIS FASTENER LOCATION IS NOT USED IF YOU MAKE THE OPTIONAL DAMAGE CUTOUT.

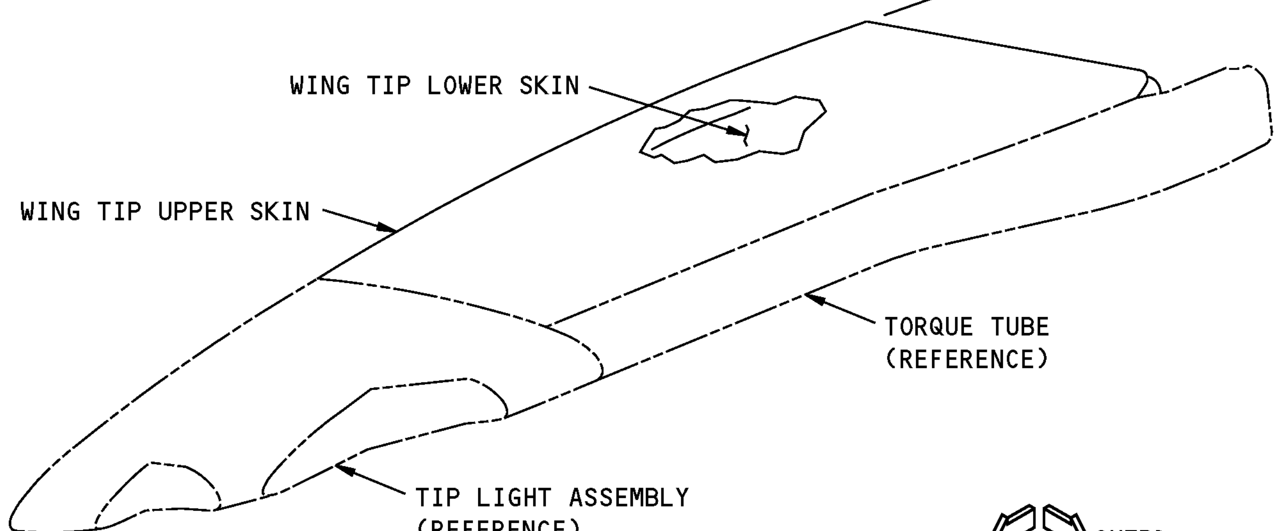
**Wing Tip Skin - External Repair**
**Figure 202**

## REPAIR 2 - WING TIP SKIN FLUSH REPAIR

### 1. Applicability

A. Repair 2 is applicable to damage to the wing tip skin shown in Wing Tip Skin Location, Figure 201/REPAIR 2.




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

(A)

**Wing Tip Skin Location**
**Figure 201**

## 2. General

A. Repair 2 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-07 | MACHINING AND DRILLING OF COMPOSITE STRUCTURES |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| AMM 57-21-11/401 | Wing Tip Removal/Installation |

## 4. Repair Instructions

A. Remove the wing tip if necessary as given in AMM 57-21-11/401.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make a minimum corner radius of 0.50 inch.

(2) Make sure the cut out hole removes all of the damaged skin.

(a) Refer to Wing Tip Skin - Flush Repair, Figure 202/REPAIR 2 for the shape of the cutout.

(3) Make sure there is a minimum of two rows of repair fasteners around the edge of the cut out.

C. Make the repair parts as shown in Wing Tip Skin - Flush Repair, Figure 202/REPAIR 2. Refer to Table 201 .

(1) Make the contour of the part [1] filler and the part [2] doubler the same as the initial contour of the skin.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Filler | 1 | Use bare or clad 7075-T6 that is 0.063 inch thick |
| [2] | Doubler | 1 | Use bare or clad 7075-T6 that is 0.071 inch thick |

D. Assemble the repair parts and drill the fastener holes.

(1) Do not countersink the fastener holes more than 80 percent of the initial skin thickness.

(a) This will prevent a knife-edge condition of the filler and the skin.

E. Remove the part [1] filler and the part [2] doubler.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and to the bare surfaces of the initial skin.

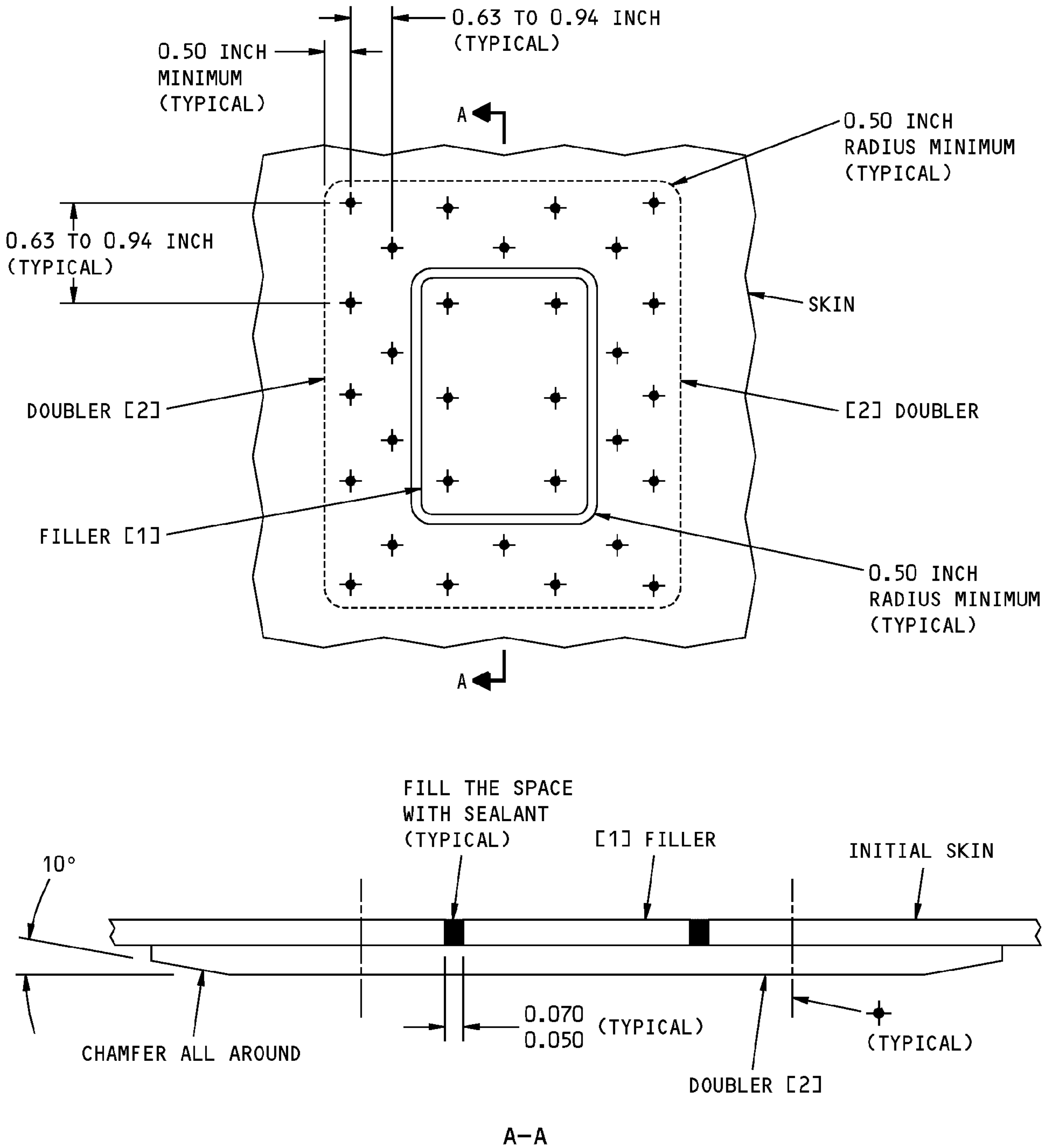
G. Apply one layer of BMS 10-79, Type II primer to the repair parts and to the edges of the skin. Refer to 51-20-01.

H. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-07.

I. Install the fasteners without sealant.

J. Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.

K. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.




FASTENER SYMBOLS

REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET

**Wing Tip Skin - Flush Repair**
**Figure 202**

## REPAIR 3 - WINGLET LEADING EDGE SKIN FLUSH REPAIR

### 1. Applicability

A. Repair 3 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Repair 3 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

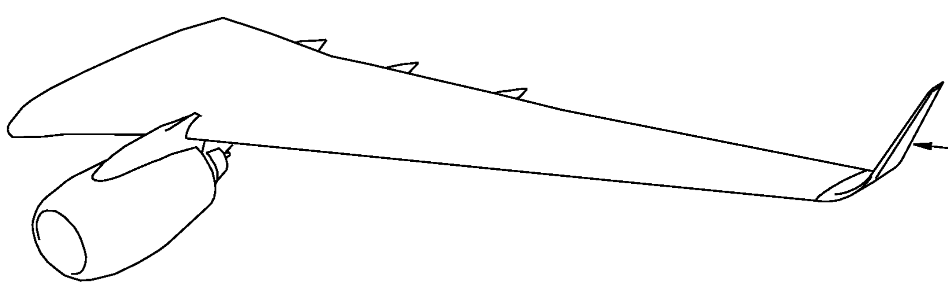
B. Repair 3 is applicable to damage to the winglet leading edge skin shown in Figure 201.

**NOTE**: A maximum of (2) winglet leading edge skin repairs is permitted from rib 27 to midspan of the outboard leading edge panel.

C. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

### 2. General

A. Repair 3 gives instructions for a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Winglet Skin Location**
**Figure 201**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| 737 NDT Part 6, 51-00-00 | Structures - General |

## 4. Repair Instructions

A. Cut and remove the damaged part of the leading edge skin. Refer to 51-10-02.

(1) Make a minimum corner radius of 0.50 inch (12.70 mm).

B. Put the skin around the cut back to the initial contour.

C. Do a high frequency eddy current (HFEC) inspection of the damaged area to make sure that all the damage is removed. Refer to 737 NDT Part 6, 51-00-00, Figure 4 for the HFEC inspection procedures.

(1) The dye penetrant inspection is permitted as an alternative to the HFEC inspection. Refer to SOPM 20-20-02.

D. Make the repair parts and make the shape the same as the initial contour of the skin. Refer to Table 201/REPAIR 3 and Figure 202.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| PART NUMBER | PART | QUANTITY | MATERIAL |
| [1] | Repair Plate | 1 | Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming. |
| [2] | Doubler | 1 | Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming. |
| [3] | Doubler | 1 | Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming. |
| [4] | Strap | 1 | Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming. |

E. Put the repair parts in place as shown in Figure 202/REPAIR 3.

F. Drill and countersink the necessary fastener holes

**NOTE**: Do not countersink more than 80 percent of the skin thickness to prevent knife-edging of the skin.

G. Remove the repair parts.

H. Remove all the nicks, scratches, burrs, gouges, and sharp edges from the repair parts and the bare surfaces of the skin.

I. Apply a chemical conversion coating to the repair parts and the bare surfaces of the initial skin. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01

J. Apply one layer of BMS 10-11, Type I primer to the initial surfaces and to the repair parts. Refer to SOPM 20-41-02.

K. Install the repair parts [2], [3], and [4]. Refer to Figure 202/REPAIR 3.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin and to the repair parts. Refer to 51-20-05.

L. Install the aluminum fasteners without sealant.

M. Install repair part [1]. Refer to Figure 202/REPAIR 3.

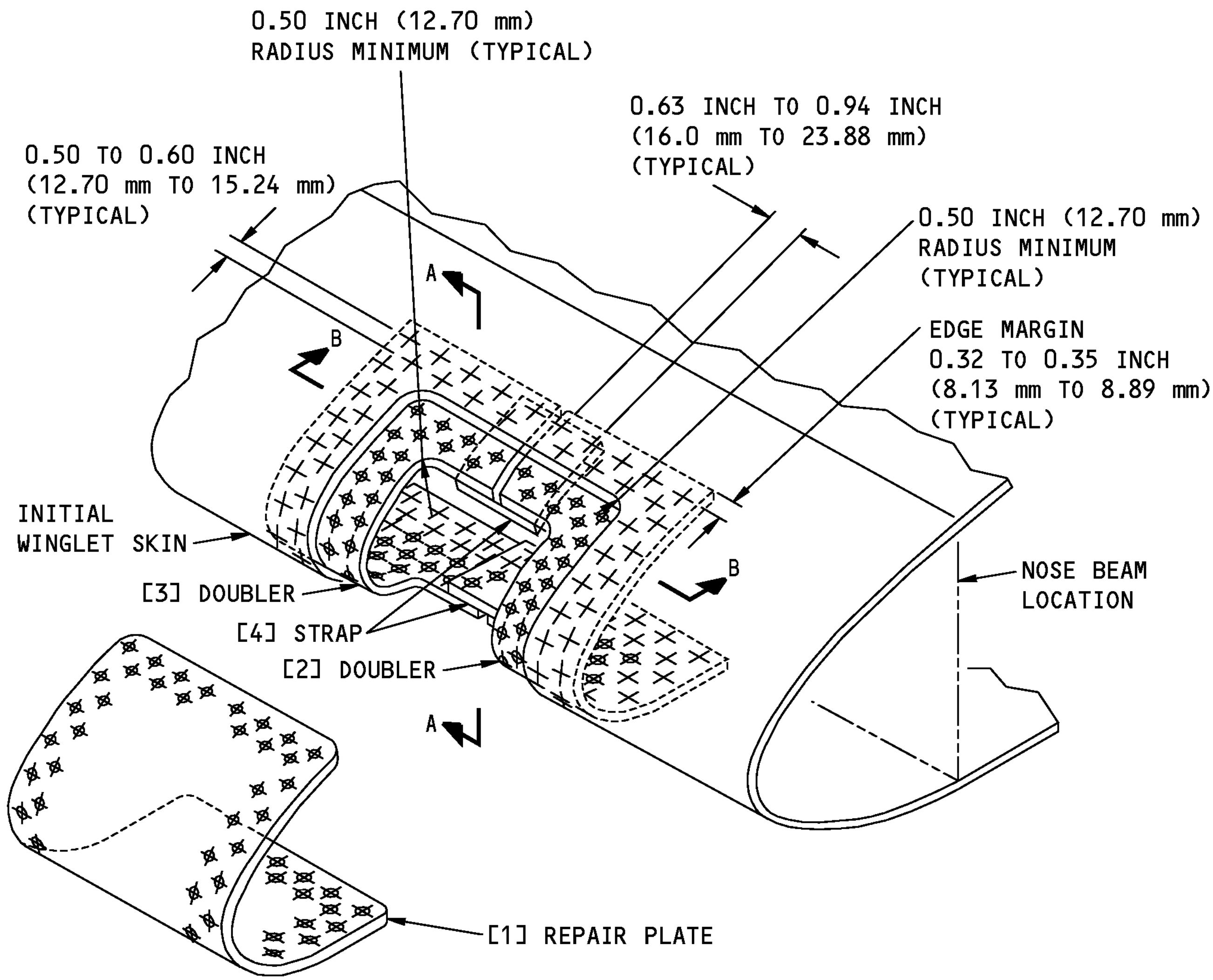
(1) Apply BMS 5-95 sealant to the mating surfaces of the repair parts. Refer to 51-20-05.

N. Install the blind fasteners with BMS 5-95 sealant.

O. Fill the gap between the skin and the part [1] repair plate with BMS 5-95 sealant. Refer to 51-20-05.

P. Apply the finish to the repair area as given in AMM 51-21-00/701.

**NOTE**: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

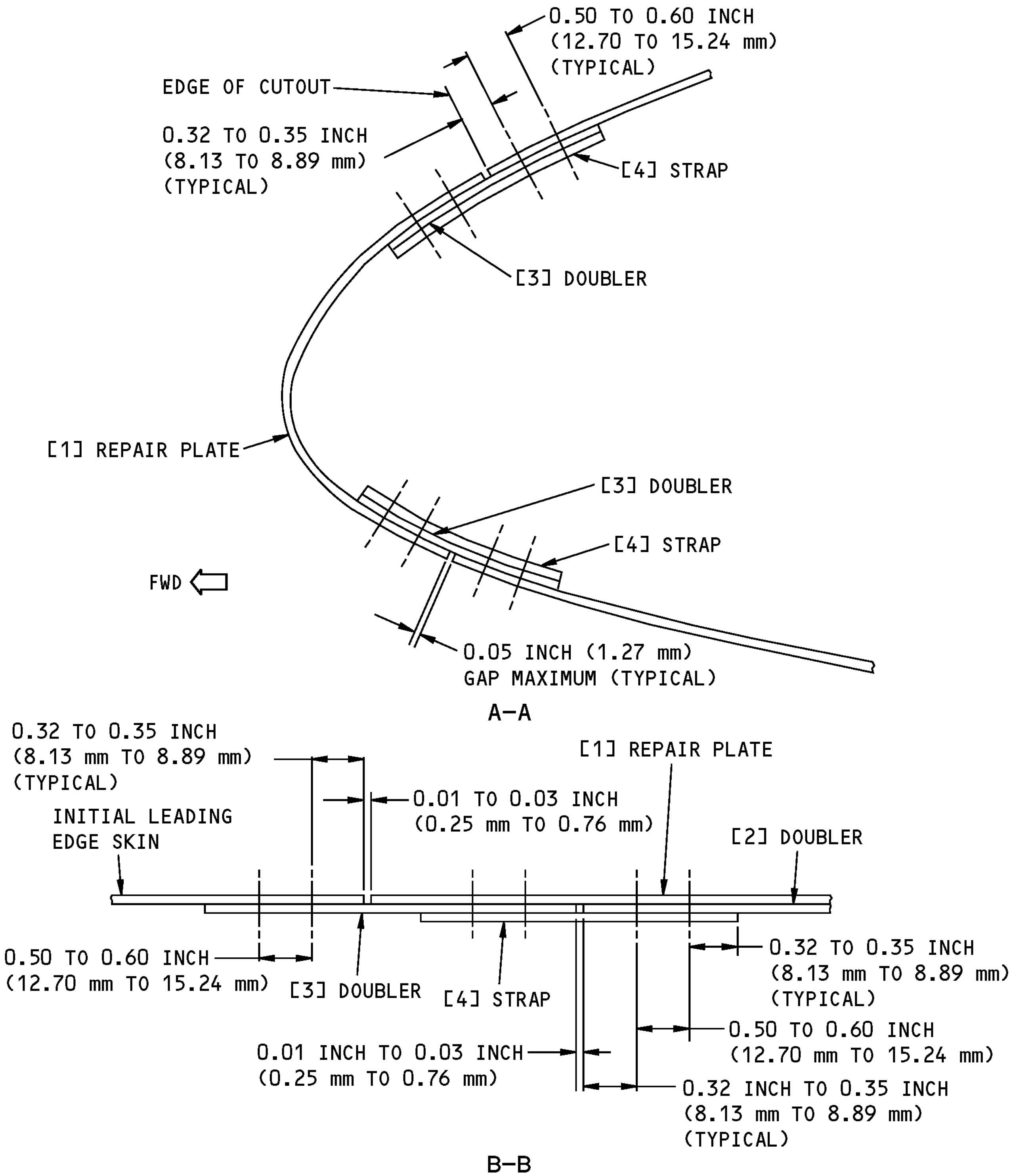



FASTENER SYMBOLS

+ REPAIR FASTENER LOCATION. INSTALL A BACR15CE5AD RIVET

⌖ REPAIR FASTENER LOCATION. INSTALL A NAS1739E5 BLIND RIVET

**Leading Edge Skin Flush Repair**
**Figure 202 (Sheet 1 of 2)**




**Leading Edge Skin Flush Repair**
**Figure 202 (Sheet 2 of 2)**

## REPAIR 4 - WINGLET SKIN REPAIR

### 1. Applicability

A. Repair 4 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Repair 4 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

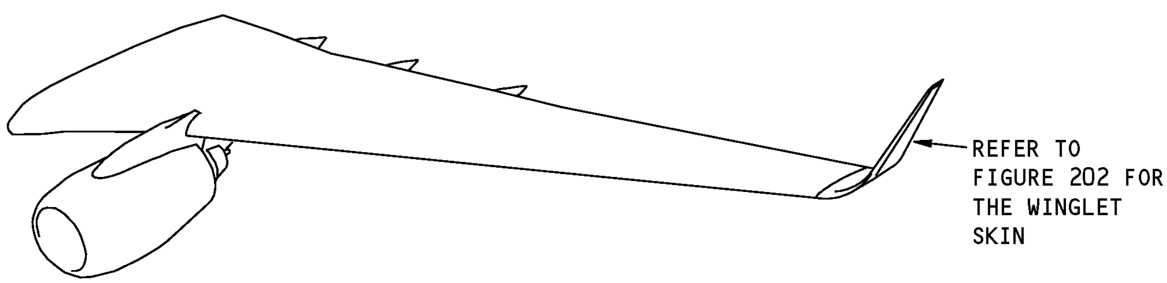
(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

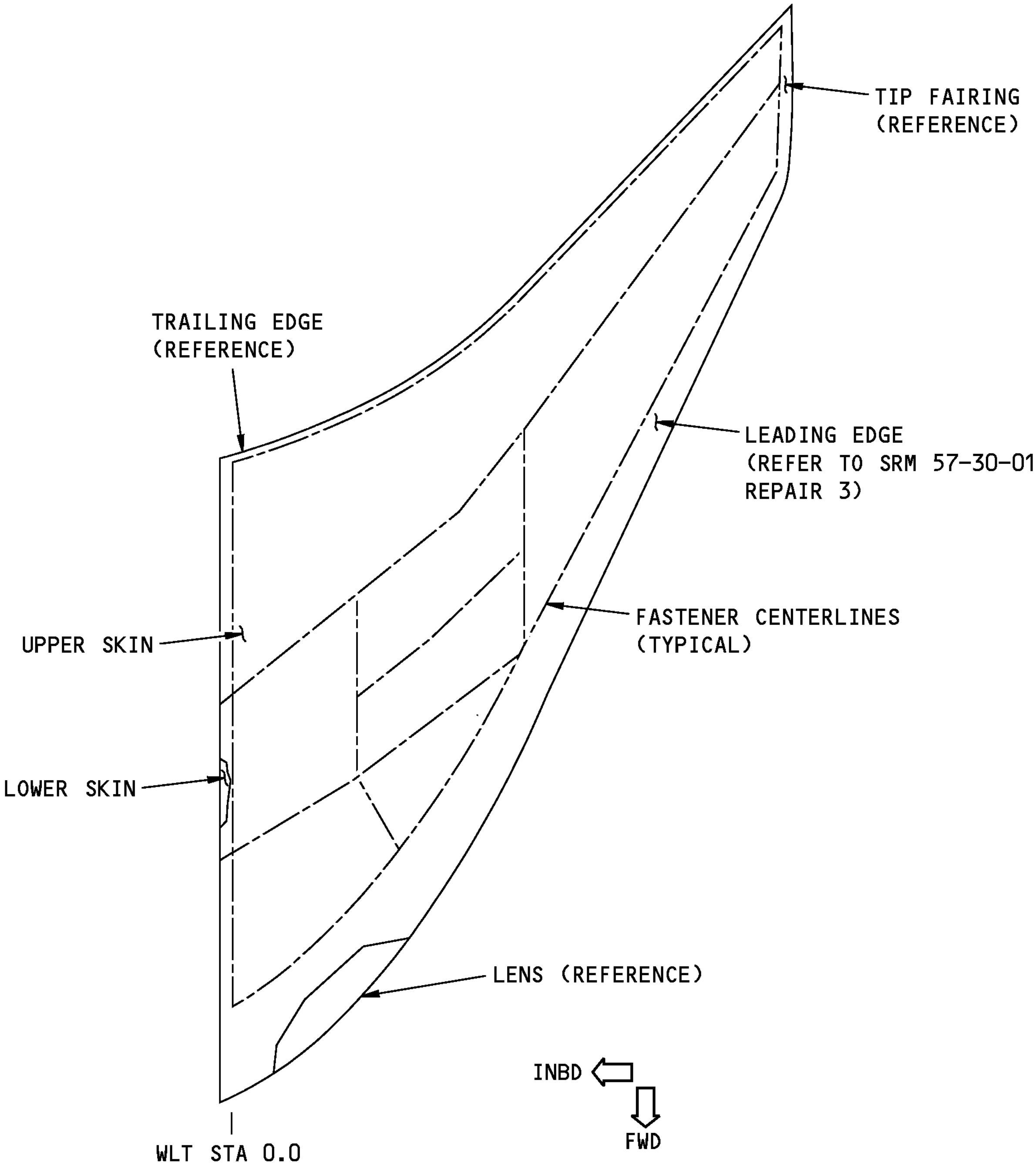
(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. This subject gives repair limits for the winglet skins shown in Figure 201 .




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Winglet Skin Location**
**Figure 201**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Winglet Skin Repair**
**Figure 202**

## 2. General

A. A maximum of (6) 5.0 inches diameter wet layup repairs and (1) 12.0 inches diameter pre-preg repair is permitted. Contact The Boeing Company for repairs more than these limits.

B. Repair 4 gives instructions for Category A repairs to all Zone locations. Refer to 51-00-06 to find the definitions of the different categories of repairs.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Nondestructive Inspection (NDI) procedure. Refer to NDT, Part 1, 51-01-02 for inspection procedures.

(1) Step sanding is not permitted.

(2) Refer to Figure 203 for the definitions of the diameter and depth of damage.

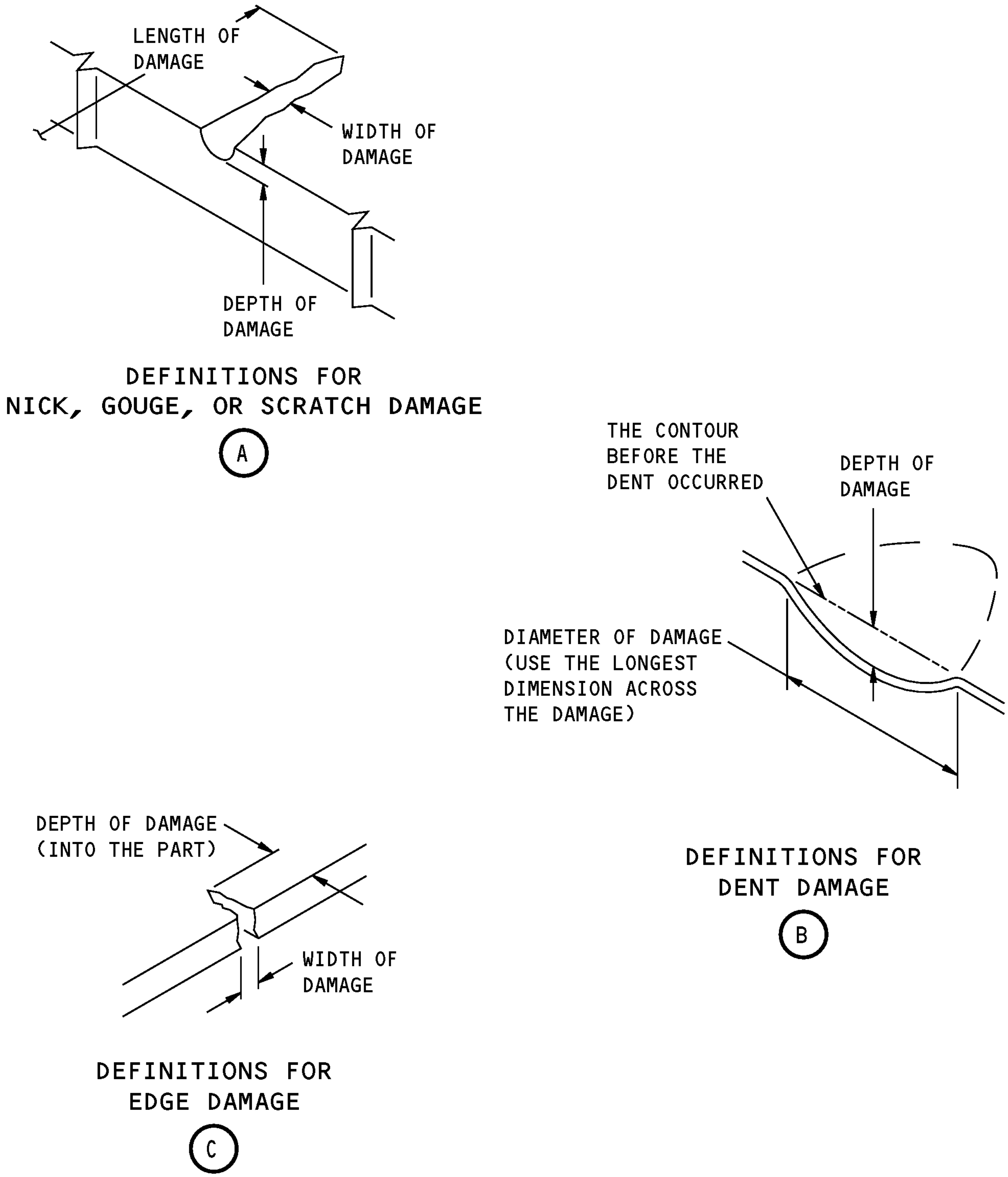
(3) Refer to Figure 204 for the definitions of the facesheets of a honeycomb core area.

D. Do the repair as given in Paragraph 4./REPAIR 4 for all Zone areas.

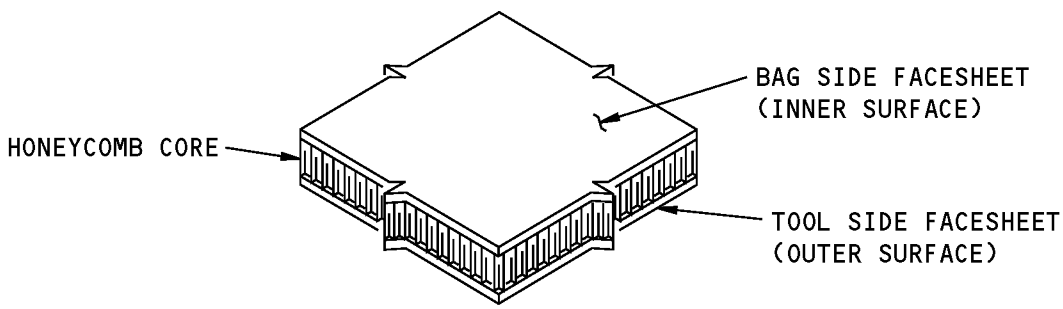
E. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

F. Refer to 51-20-01 for interior and exterior finishes.

**NOTE**: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.




**Damage Definitions**
**Figure 203**




**Definitions of the Facesheets**
**Figure 204**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |

4. **Repair Instructions**

A. For cracks, holes, delamination, nicks and gouges, refer to Table 201/REPAIR 4 for the repair limits and procedures.

B. For repairs made with wet layup materials, do as follows:

(1) For each facesheet that is damaged, use one structural repair fabric ply for each initial fabric ply that was damaged. Use two structural repair fabric plies for each initial tape ply that was damaged.

(2) Then add two more structural repair plies. Put one structural ply at 0 or 90 degrees and put the next ply at +/- 45 degrees to the core ribbon direction.

**NOTE**: A maximum of (6) 5.0 inches diameter repairs are permitted.

C. For repairs made with preimpregnated layup materials, do as follows:

(1) For each facesheet that is damaged, use one structural repair ply for each initial ply that was damaged. The type, class, grade, and style shall be the same as the initial plies.

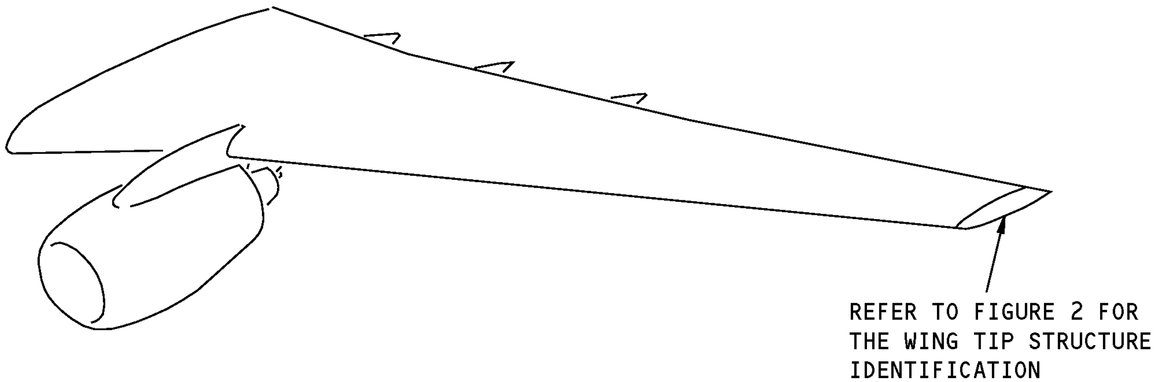
(2) Then add two more structural repair plies. Put one structural ply at 0 or 90 degrees and put the next ply at +/- 45 degrees to the core ribbon direction.

**NOTE**: A maximum of (1) 12.0 inches diameter repair is permitted.

**Table 201**:

| REPAIR DATA FOR THE 350°F (177°C) CURE, WINGLET FIBERGLASS/EPOXY SKIN PANELS | | | |
|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY A WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **350°F (177°C)** |
| REPAIR SIZE | - This temperature repair is not permitted. | Damage that is a maximum of:<br>- 5.0 inches in diameter<br>A maximum of (6) repair locations is permitted. Contact The Boeing Company for repairs more than these limits<br>6.0 inches minimum clearance from other damage | Damage that is a maximum of:<br>- 12.0 inches in diameter<br>A maximum of (1) repair location is permitted. Contact The Boeing Company for repairs more than these limits. |
| REPAIR INSTRUCTIONS | ———— | SRM 51-70-04 and Paragraph 4.B | SRM 51-70-05 and Paragraph 4.C |

## IDENTIFICATION 1 - WING TIP STRUCTURE



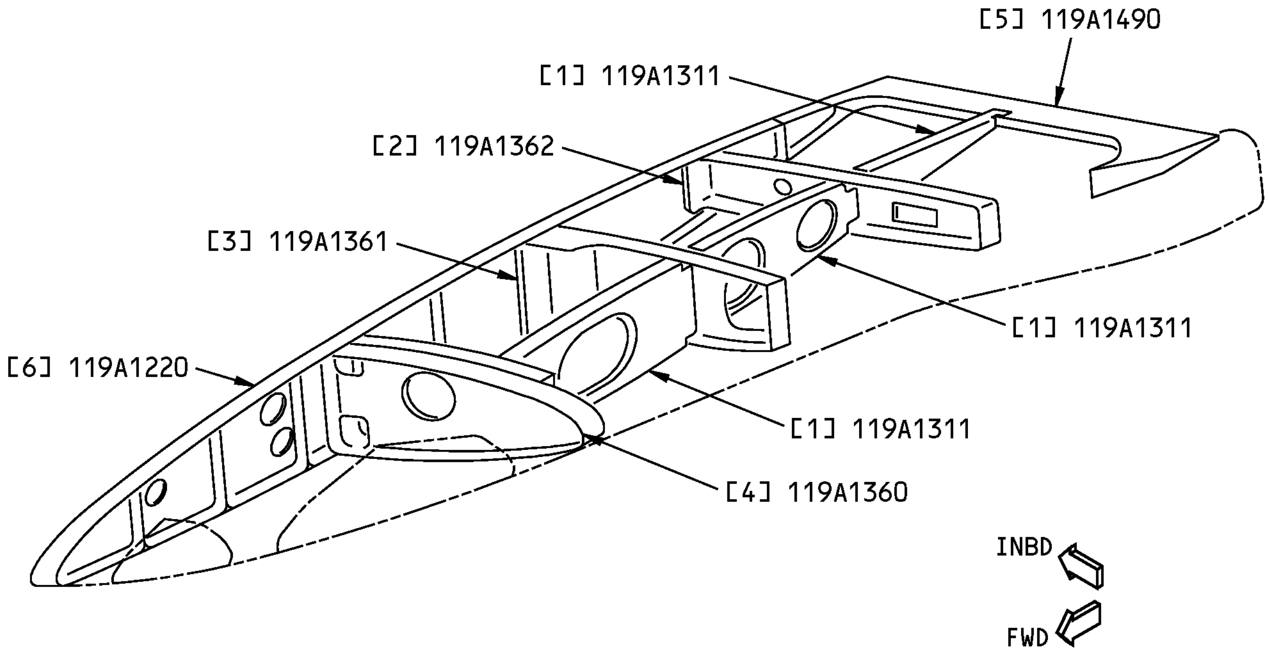

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Tip Structure Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 119A1200 | Wing Tip Installation |
| 119A1500 | Wing Tip Fairing and Light Assembly |
| 119A0100 | Wing Tip Assembly |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Tip Structure Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Intercostal | 0.040 (1.02) | 7075-T6 clad sheet | |
| [2] | Aft Rib Former | | 7050-T7451 plate as given in AMS 4050 | |
| [3] | Mid Rib Former | | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Forward Rib Former | | 7050-T7451 plate as given in AMS 4050 | |
| [5] | Trailing Edge Wedge | | A357.0-T6 as given in MIL-A-21180, Class 10 | |
| [6] | Bulkhead Rib | | 7050-T7451 plate as given in AMS 4050 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - WINGLET STRUCTURE

### 1. Applicability

A. Identification 2 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. Refer to Table 1 for the reference drawings and Table 2 for the List of Materials.

**Table 1**:

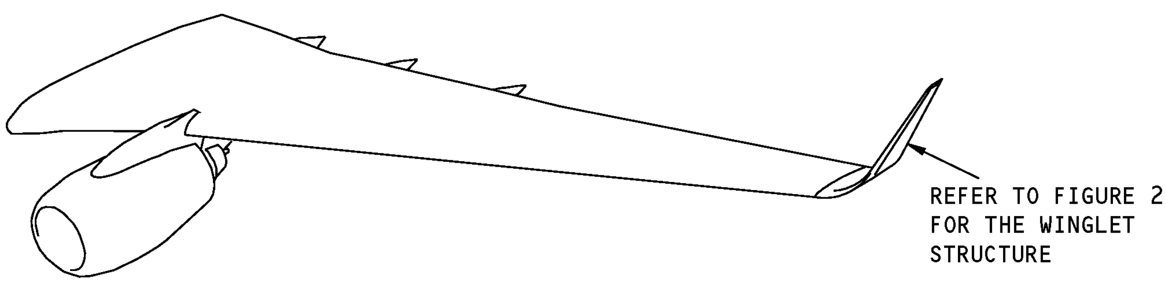
| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 737-0002 | Winglet Installation |
| 737-0010 | Winglet Final Assembly |
| 737-0011 | Winglet Assembly Substructure |

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Trailing Edge Rib Sta 0 | | 7050-T7451 as given in BMS 7-323 | |
| [2] | Inspar Rib Sta 0 | | 7050-T7451 as given in BMS 7-323 | |
| [3] | Inspar Rib Sta 1 | | 7050-T7451 as given in BMS 7-323 | |
| [4] | Inspar Rib Sta 4 | | 7050-T7451 as given in BMS 7-323 | |
| [5] | Leading Edge Rib Sta 0 | | 7050-T7451 as given in BMS 7-323 | |
| [6] | Leading Edge Rib Sta 1 | | 7050-T7451 as given in BMS 7-323 | |
| [7] | Leading Edge Rib Sta 4 | | 7050-T7451 as given in BMS 7-323 | |
| [8] | Bonded Rear Spar | | Fiberglass/Graphite/Epoxy Solid Laminate<br>Graphite/Epoxy Fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW<br>Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190<br>Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1<br>Refer to APB drawing 737-0241 | |
| [9] | Bonded Center Spar | | Fiberglass/Graphite/Epoxy Solid Laminate<br>Graphite/Epoxy Fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | | | Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190<br>Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1<br>Refer to APB drawing 737-0231 | |
| [10]<br><br>[10] | Bonded Front Spar | | Fiberglass/Graphite/Epoxy Solid Laminate<br>Graphite/Epoxy Fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW<br>Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190<br>Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1<br>Refer to APB drawing 737-0221 | |
| [11] | Lower Stiffener | | 7075-T73 bare sheet as given in QQ-A-250/12 | |
| [12] | Upper Stiffener | | 7075-T73 bare sheet as given in QQ-A-250/12 | |
| [13] | Lower Splice Plate | | Ti-6Al-4V as given in MIL-T9046, AB-1 condition | |
| [14] | Upper Splice Plate | | Ti-6Al-4V as given in MIL-T-9046, AB-1 condition | |
| [15] | Attach Clip | | 2024-T3 clad sheet as given in QQ-A-250/5 | |

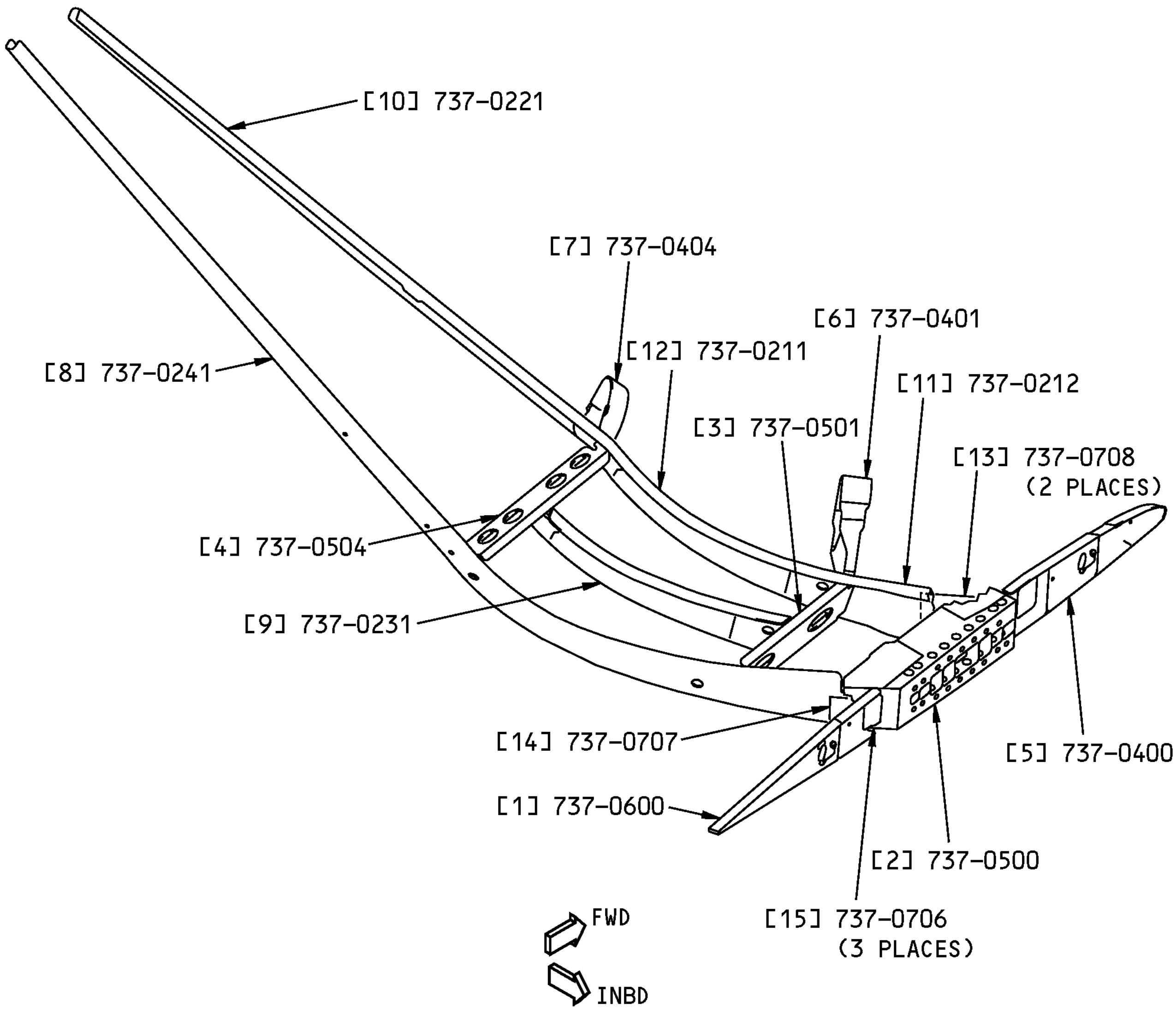
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Winglet Structure Location**
**Figure 1**




LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

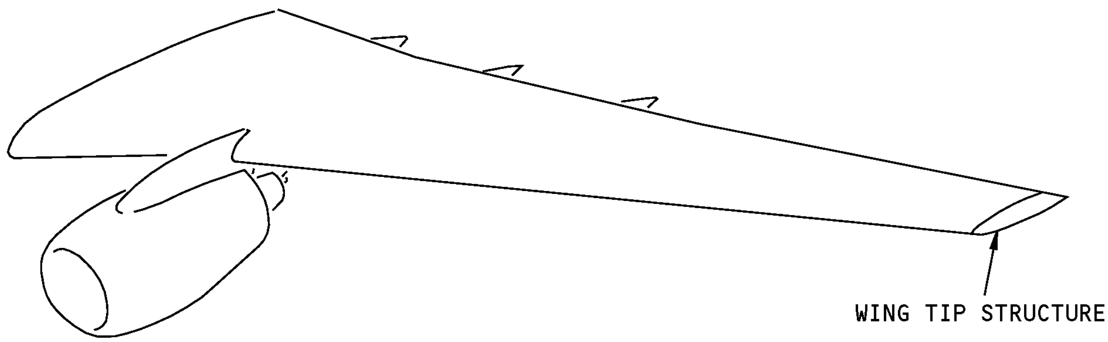
NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Winglet Structure Identification**
**Figure 2**

## ALLOWABLE DAMAGE 1 - WING TIP STRUCTURE

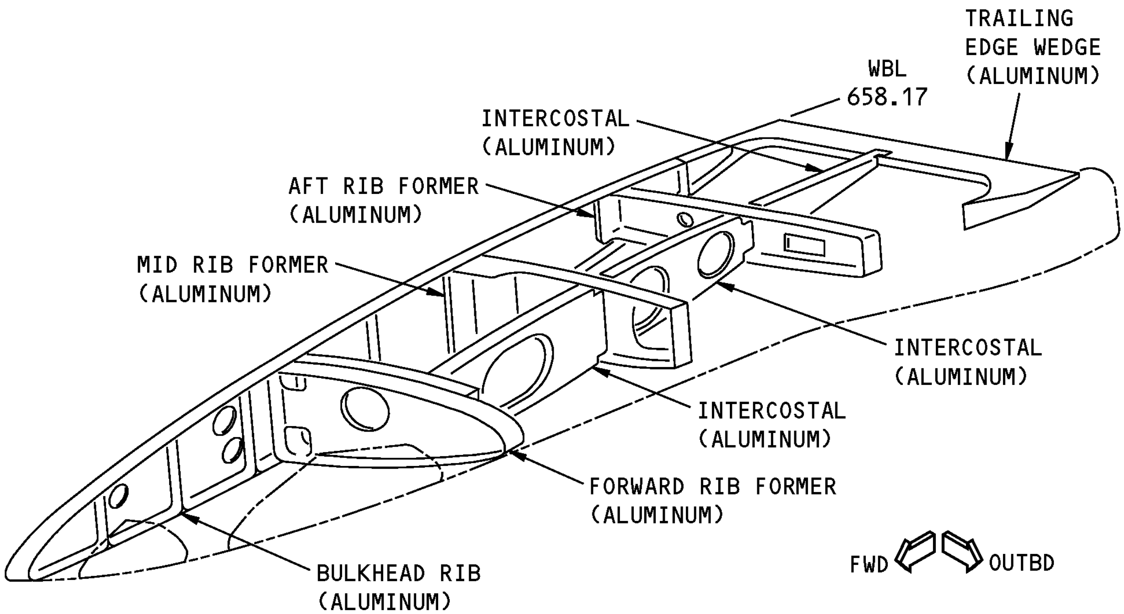
1. **Applicability**

   A. This subject gives the allowable damage limits for the wing tip structure shown in Wing Tip Structure Location, Figure 101/ALLOWABLE DAMAGE 1.




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Tip Structure Location**
**Figure 101**




**Wing Tip Structure**
**Figure 102**

2. <u>General</u>

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the ribs.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

(c) Do not flap peen or shot peen the trailing edge wedge casting or the intercostal.

(2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(3) Apply one layer of BMS 10-11, Type I primer to the reworked areas of the intercostal and ribs. Refer to SOPM 20-41-02.

(4) Apply one layer of BMS 10-79, Type III primer to the reworked areas of the trailing edge wedge. Refer to SOPM 20-44-04.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Cracks:

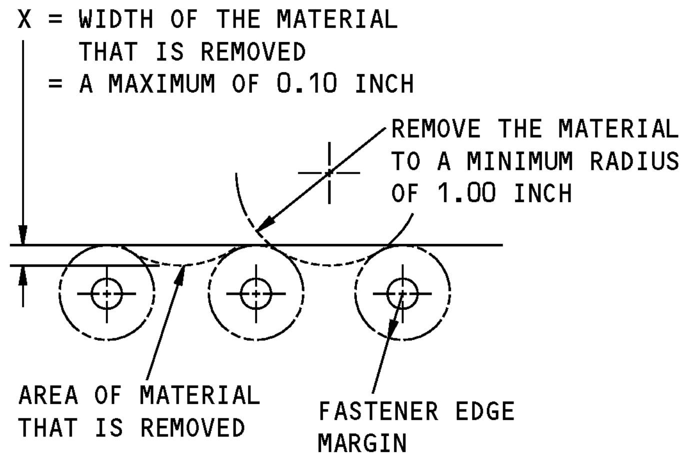
(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and F .

B. Nicks, Gouges, Scratches and Corrosion:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , E , and F .

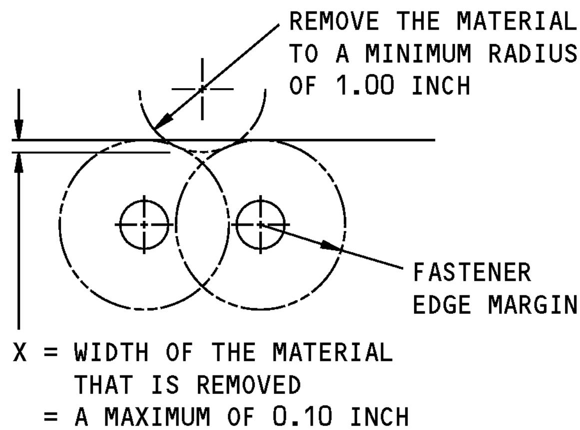
C. Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail D .

D. Holes and Punctures are not permitted.




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

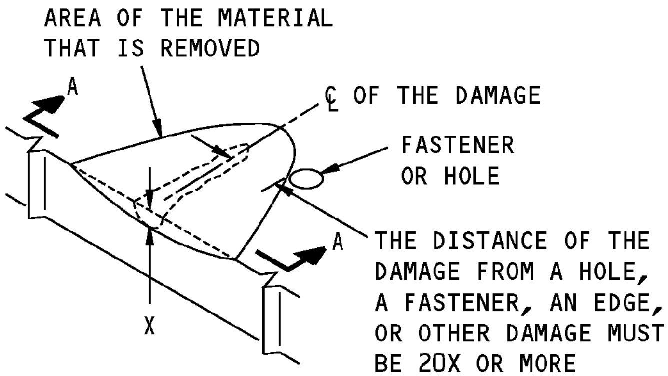
A



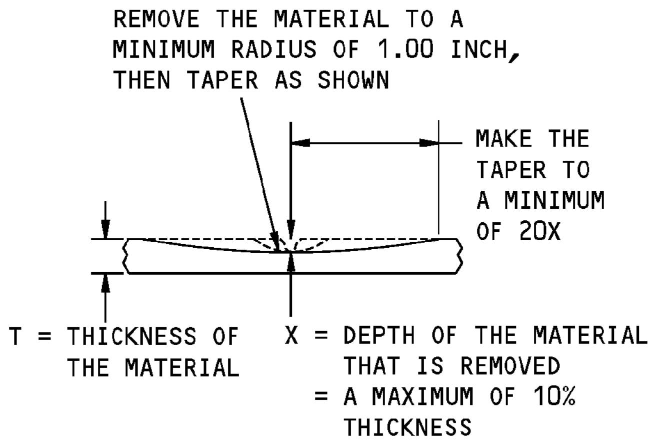

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

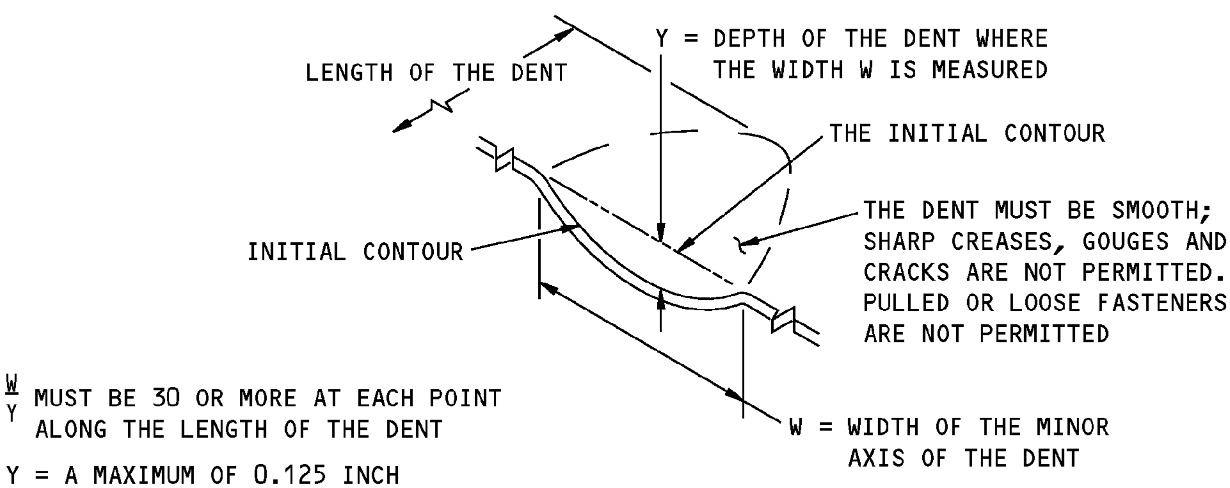
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 3)**




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE
C

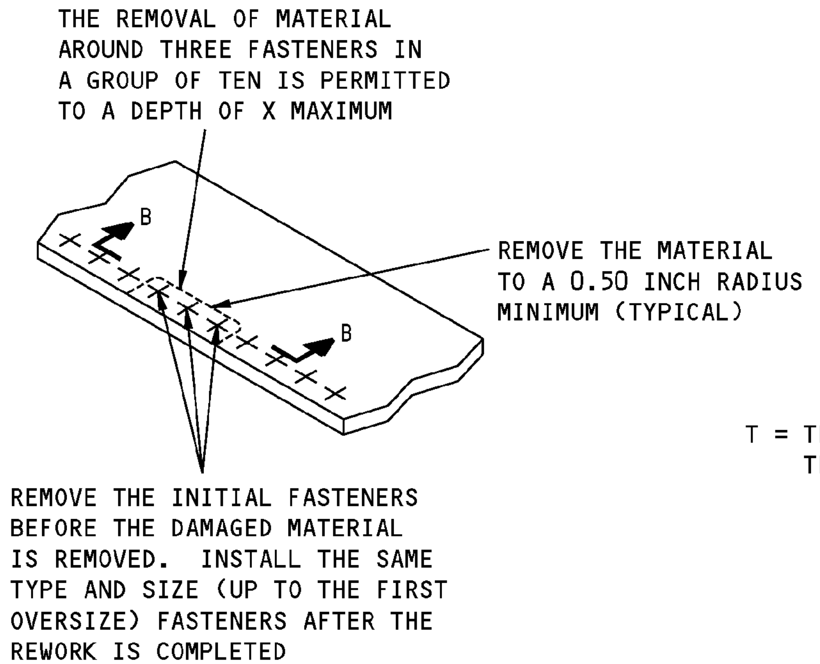



A-A



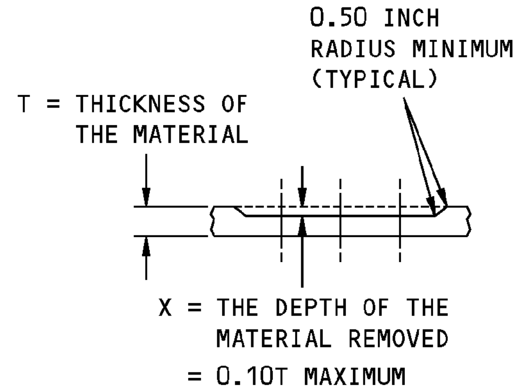

DENT THAT IS PERMITTED
D

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 3)**

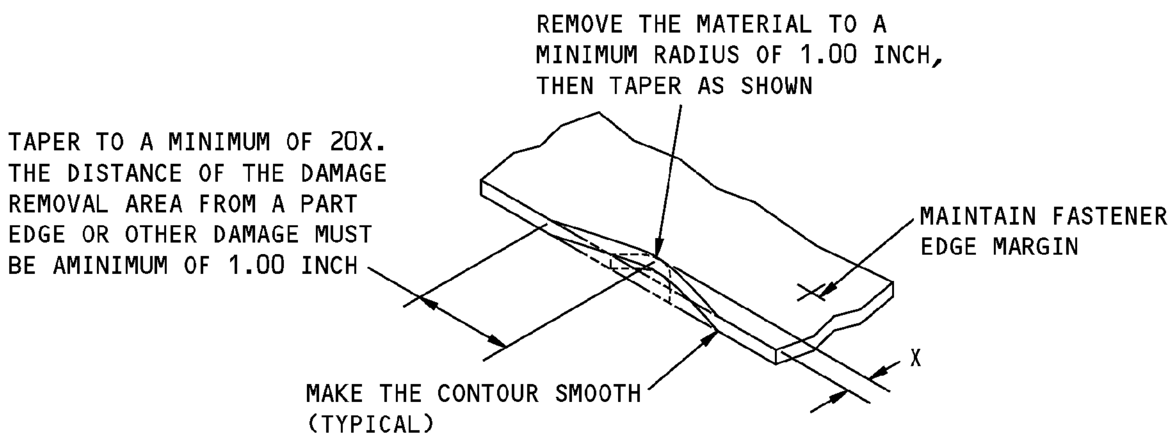



REMOVAL OF CORROSION AROUND THE FASTENERS

E




B-B




X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

F

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 3)**

## ALLOWABLE DAMAGE 2 - WINGLET STRUCTURE

### 1. Applicability

A. Allowable Damage 2 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Allowable damage 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

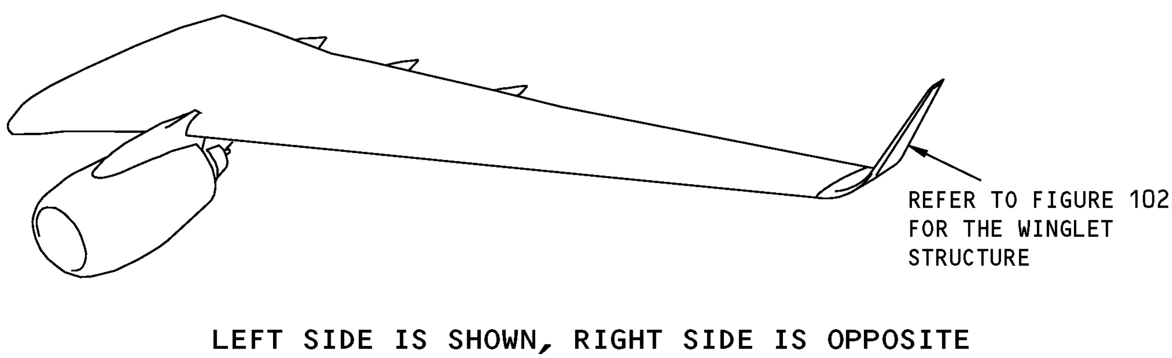
(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

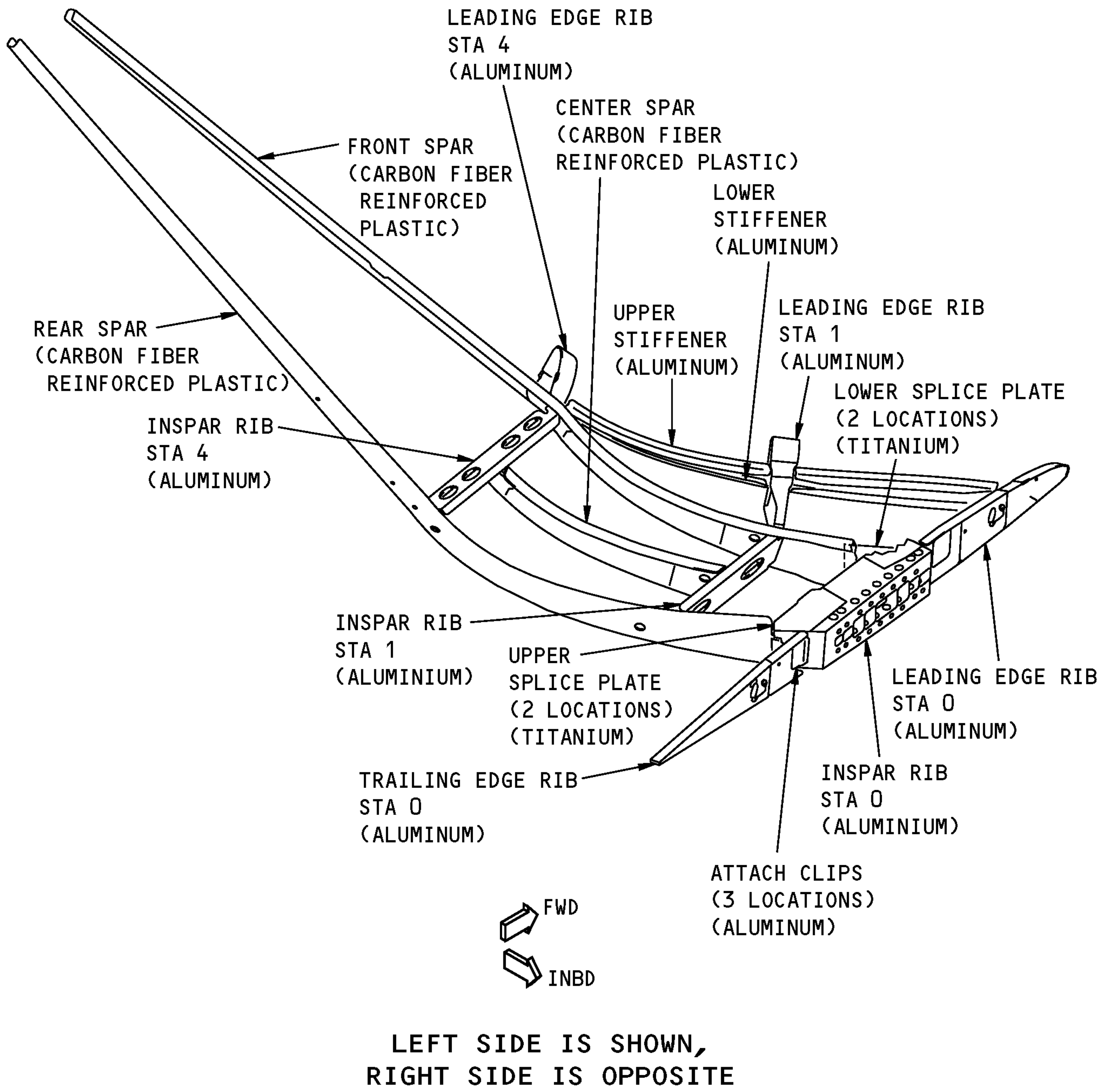
(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. This subject gives the allowable damage limits for the winglet structure shown in Figure 101.




**Winglet Structure**
**Figure 101**




**Winglet Structure**
**Figure 102**

2. **General**

A. Refer to Paragraph 4 through Paragraph 5 for the allowable damage limits.

**WARNING:** SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STREAM EXPLOSION. IF YOU DO NO OBEY, AN INJURY CAN OCCUR.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the reworked areas of the aluminum parts. Refer to 51-20-01.

(2) Apply one layer of BMS 10-11, Type I primer to the reworked areas of the aluminum and titanium parts. Refer to SOPM 20-41-02.

(3) Apply one layer of BMS 10-60, Type II primer to the reworked areas of the aluminum parts. Refer to AMM 51-21-00/701.

**NOTE**: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

D. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in economic performance of the airplane. Refer to 51-10-01.

3. **References**

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

4. **Allowable Damage Limits for the Metallic Winglet Structure**

A. Edge Cracks:

(1) Remove the damage as shown in Figure 103 , Details A , B , and C .

B. Nicks, Gouges, Scratches and Corrosion:

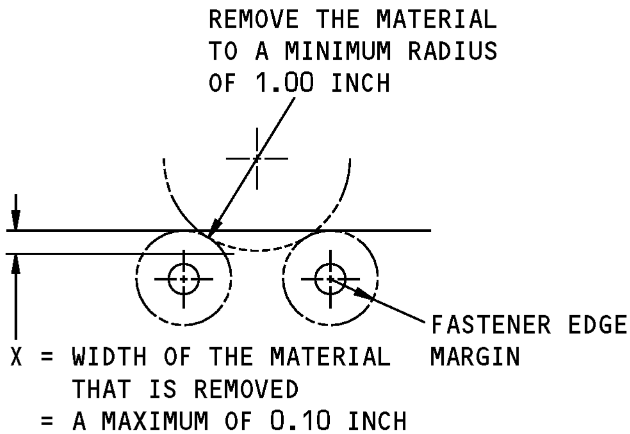
(1) Remove the damage as shown in Figure 103 , Details A , B , C , D , and E .

C. Dents are permitted as shown in Figure 103 , Detail F .

D. Holes and Punctures are not permitted.

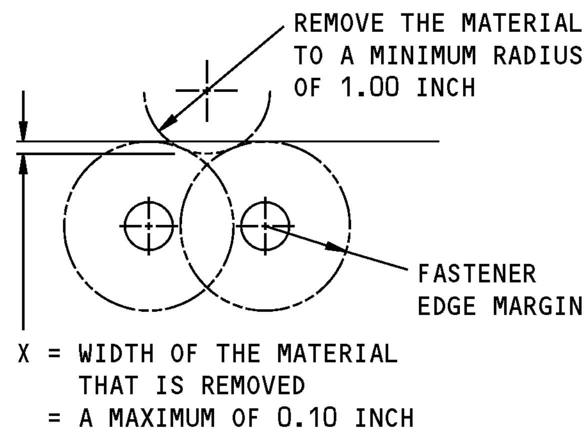
## 5. Allowable Damage Limits for the Winglet Rear, Center, and Front Spar

A. Damage is not permitted.



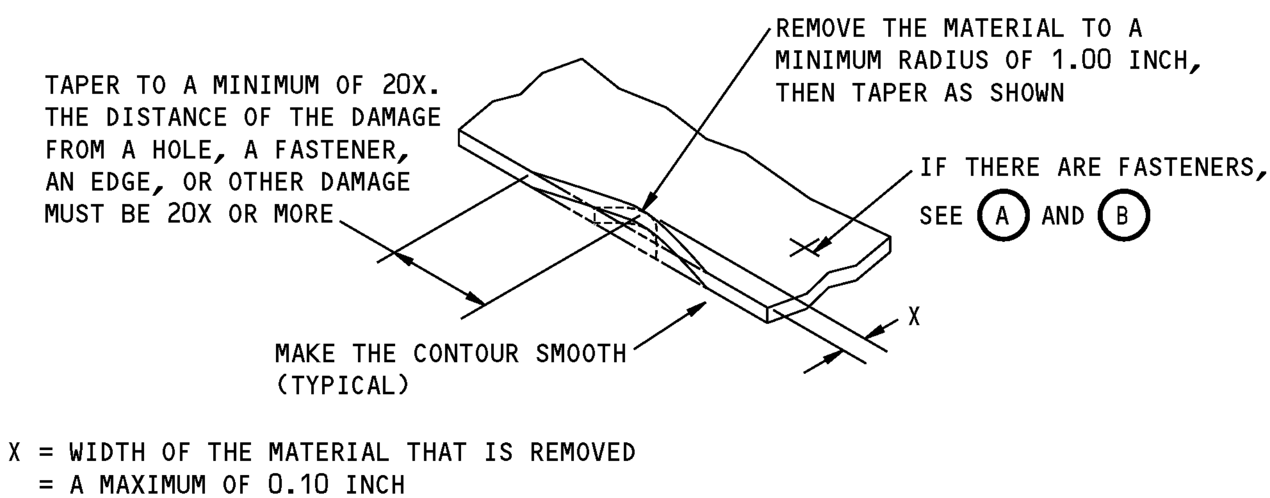

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

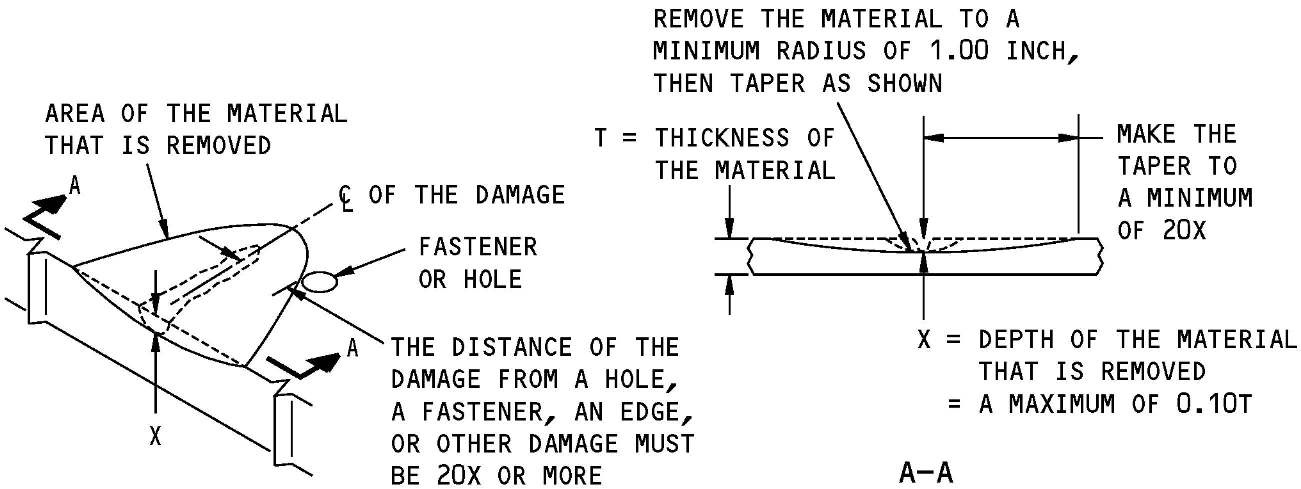
B




REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

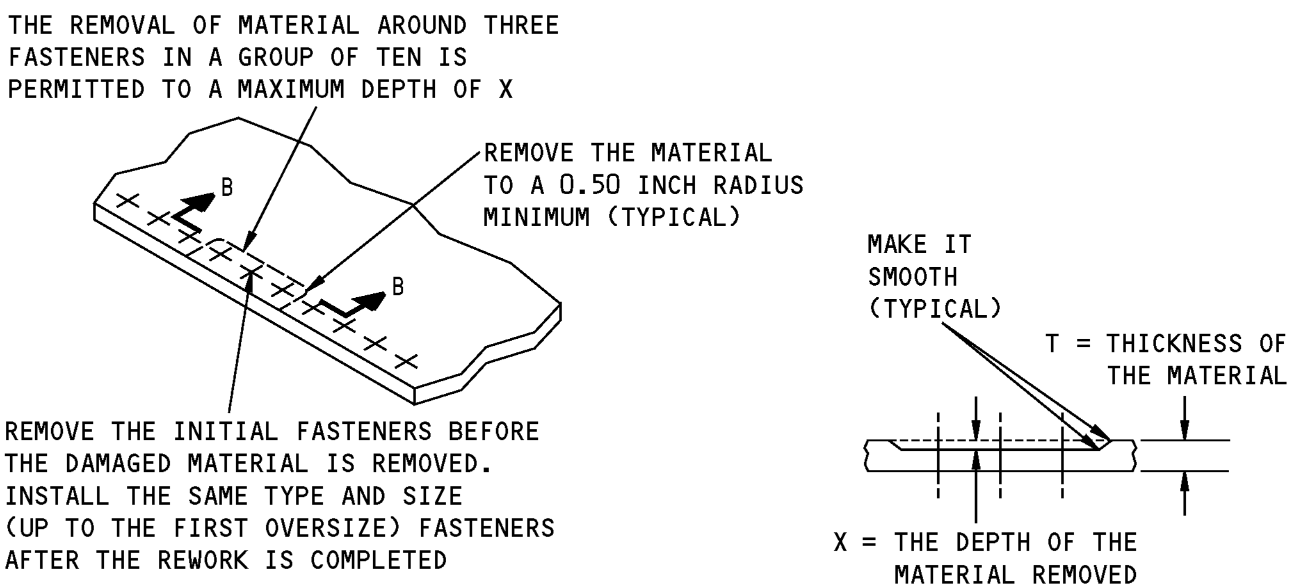
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 3)**




A-A

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

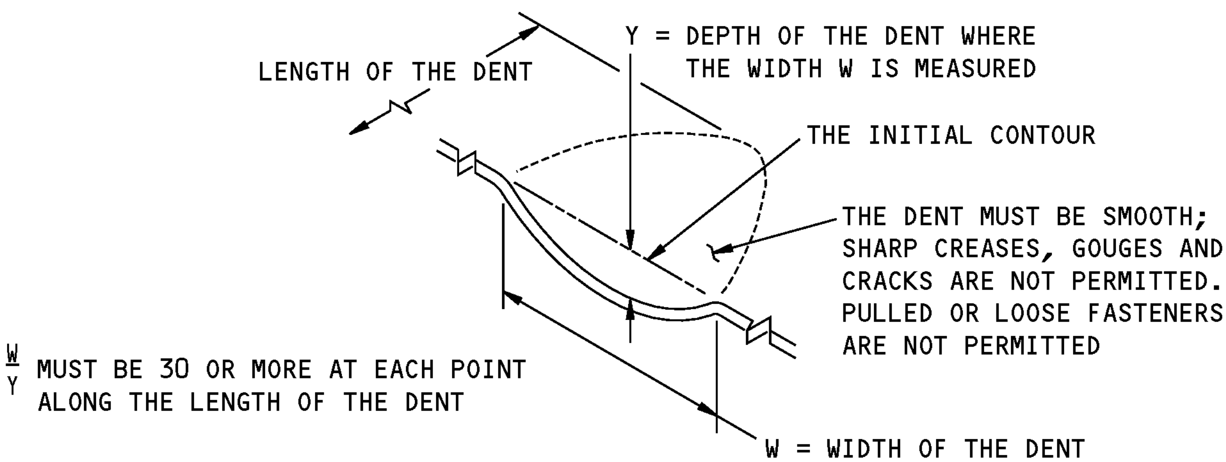



REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

E

B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 3)**

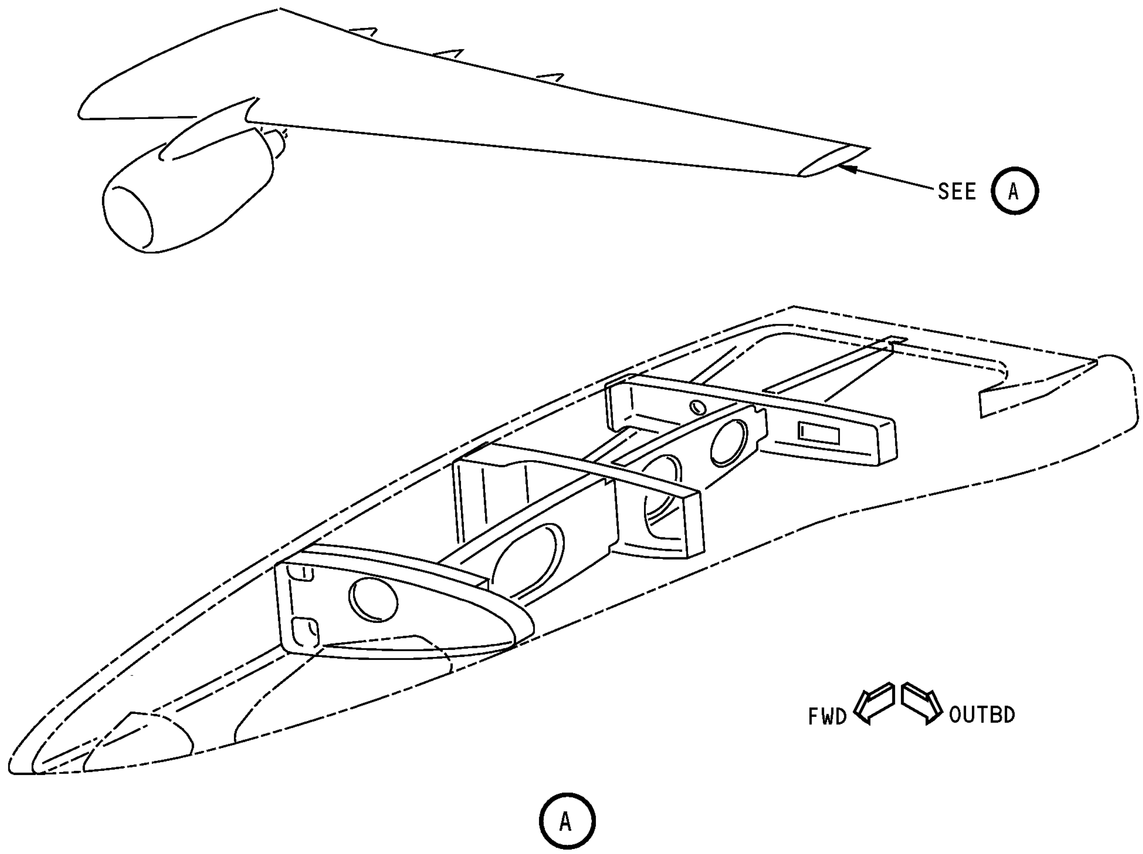



DENT THAT IS PERMITTED

F

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 3)**

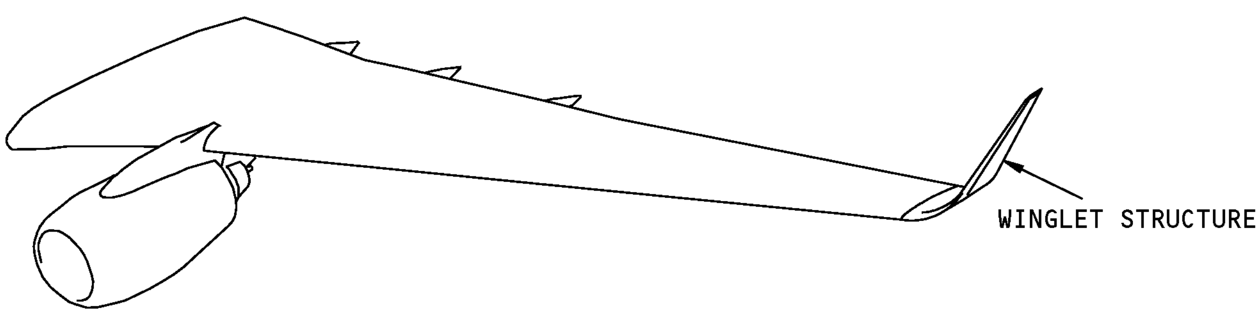
## REPAIR 1 - WING TIP STRUCTURE




NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

**Wing Tip Structure Repair**
**Figure 201**

## REPAIR 2 - WINGLET STRUCTURE

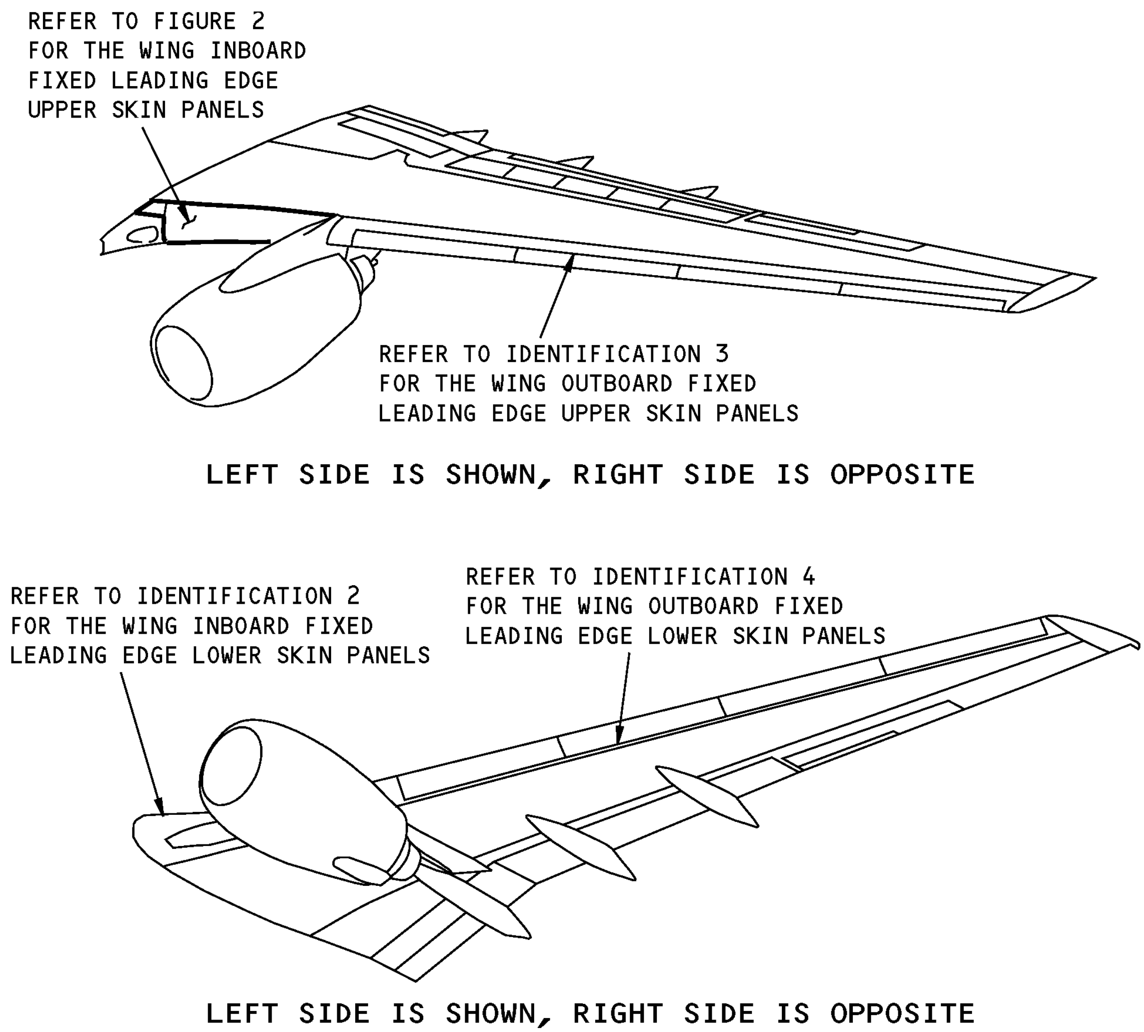



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

**Winglet Structure Repair**
**Figure 201**

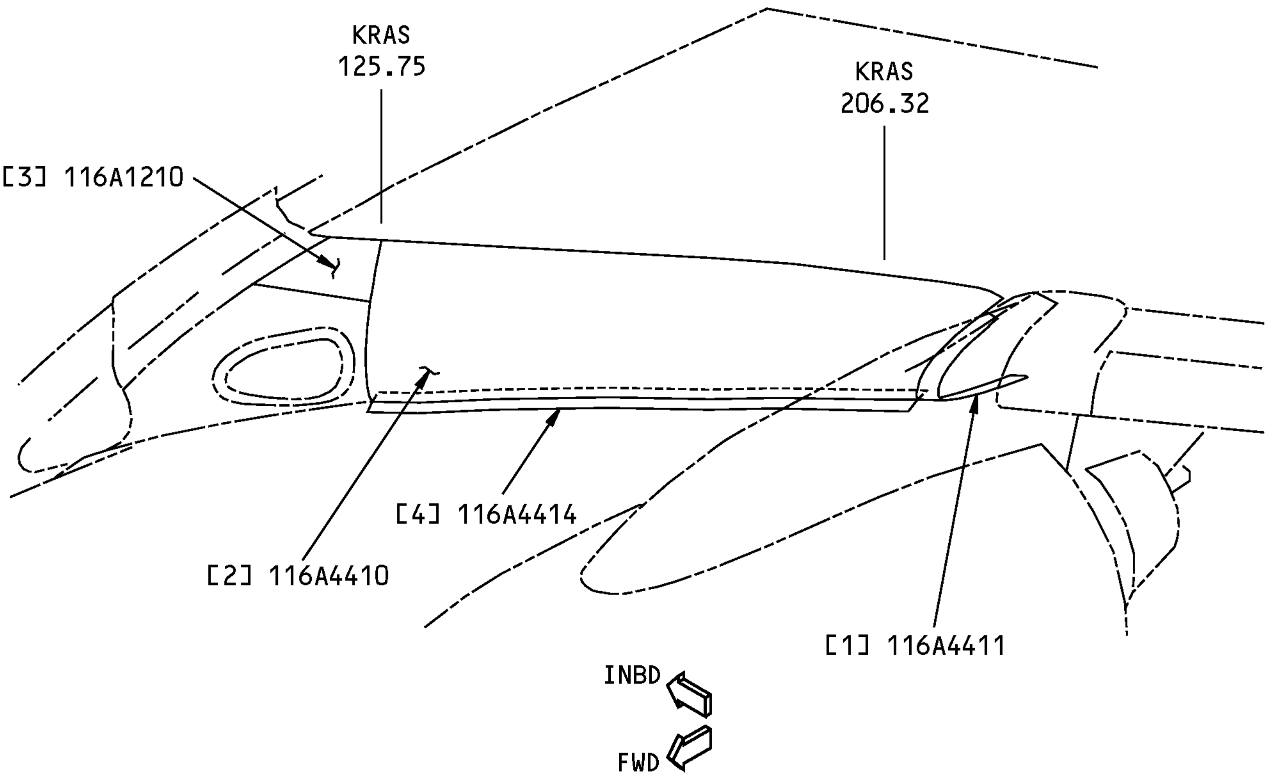
## IDENTIFICATION 1 - WING INBOARD FIXED LEADING EDGE UPPER SKIN PANELS




**Wing Inboard Fixed Leading Edge Upper Skin Panel Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A1001 | Wing Left - Product Collector |
| 116A0010 | Fixed Leading Edge Collector - Final Assembly |
| 116A0015 | Functional Product Collector - Front Spar Leading Edge Join, Left Wing |
| 116A1200 | Upper Removable Access Panel Installation, Inboard Leading Edge |
| 116A4008 | Skin Installation - Upper, Inboard Leading Edge, Wing |



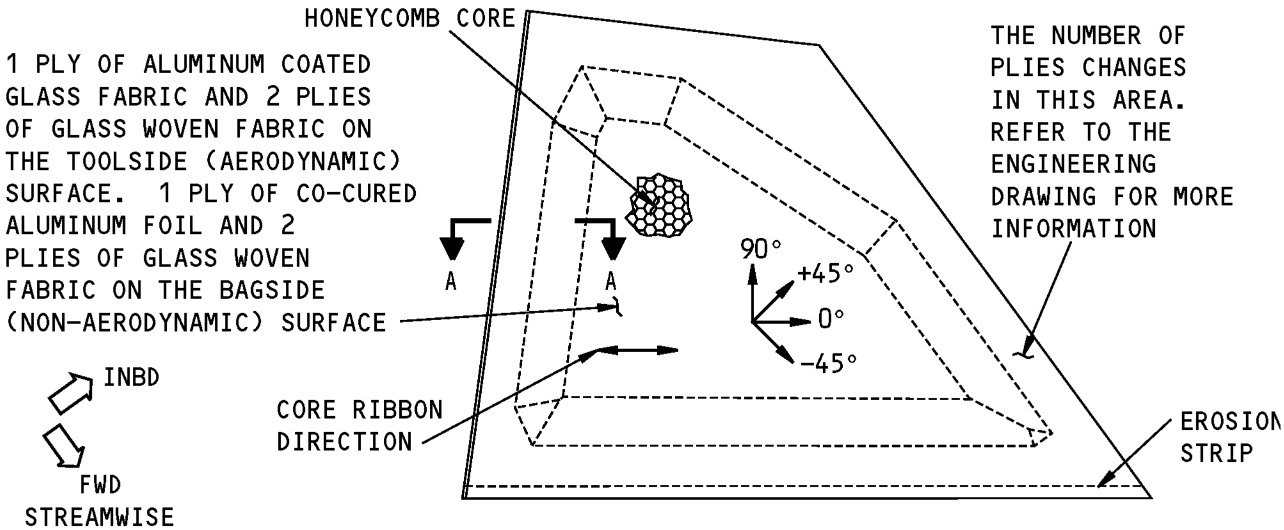

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Inboard Fixed Leading Edge Upper Skin Panel Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Skin, Closeout | 0.063 (1.6) | 2024-T42 clad sheet as given in QQ-A-250/5 | |
| [2] | Upper Skin, Strakelet to Seal Rib | 0.063 (1.6) | 2024-T42 clad sheet as given in QQ-A-250/5 | |
| [3] | Bond Assembly, Upper Removable Access Panel | | | |
| | Skin Assembly | | Refer to Figure 3 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [4] | Stall Strip | 0.50 (12.7) | 7050-T7451 machined aluminum plate as given in AMS 4050 | CUM LINE 21 AND ON |

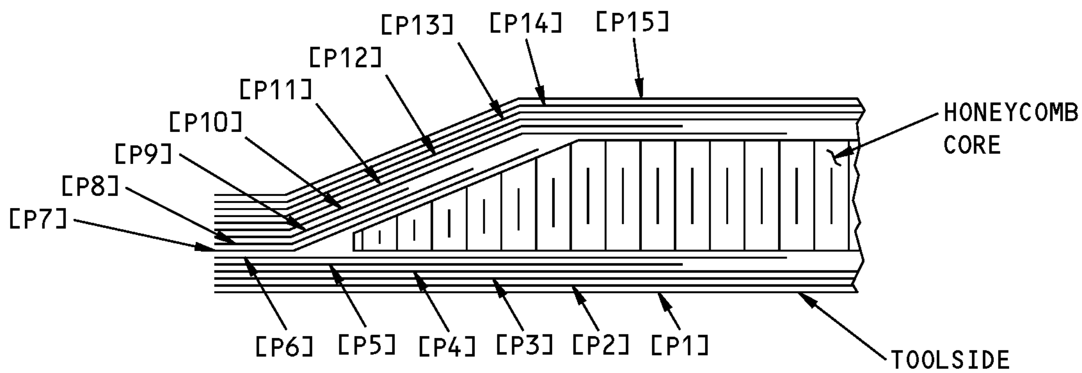
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION AND CORE RIBBON DIRECTION

A




A-A

NOTES

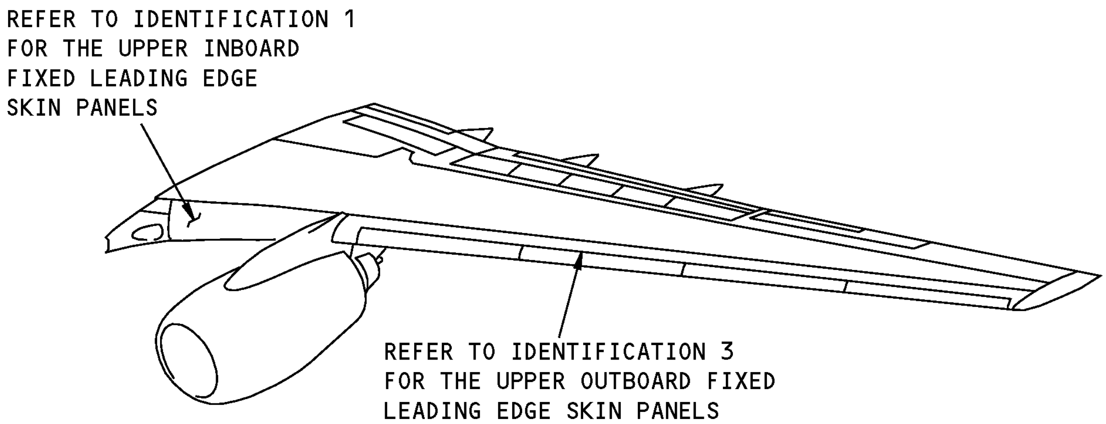
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Upper Removable Access Panel, Figure 2, Item [3]**
**Figure 3**

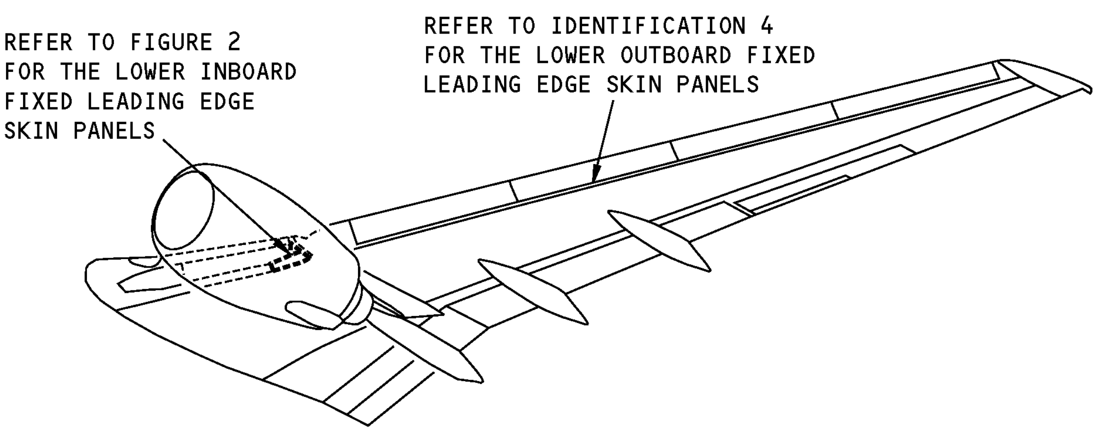
**Table 3**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [3] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A) |
| P2 to P4, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P5, P6, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |

## IDENTIFICATION 2 - WING INBOARD FIXED LEADING EDGE LOWER SKIN PANELS




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE



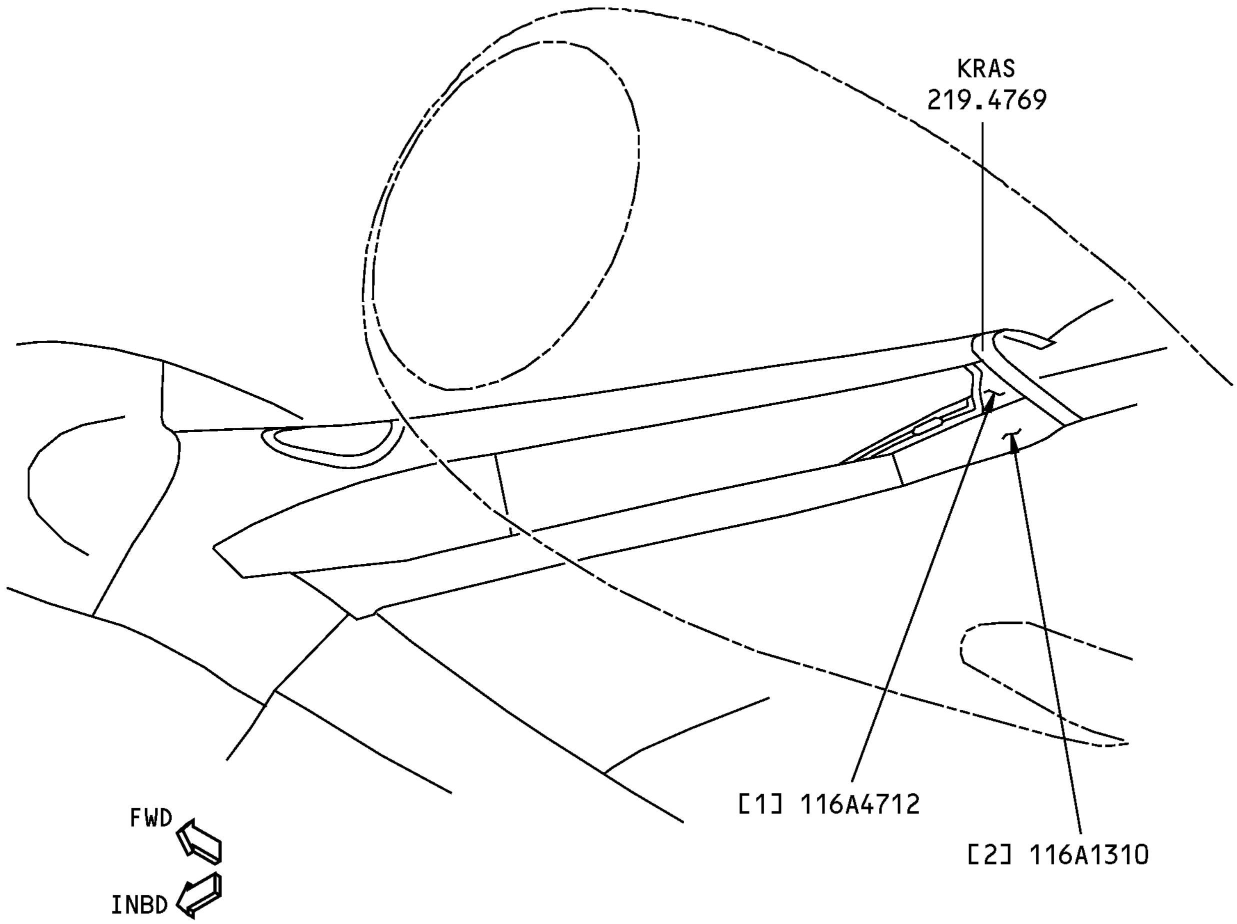

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Inboard Fixed Leading Edge Lower Skin Panel Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A1001 | Wing Left - Product Collector |
| 116A0010 | Fixed Leading Edge Collector - Final Assembly |
| 116A0011 | Fixed Leading Edge Collector - Post Paint - Left Wing |
| 116A0013 | Collector - Leading Edge, Wing Build-Up, Left Wing |
| 116A0015 | Functional Product Collector - Front Spar Leading Edge Join, Left Wing |
| 116A1300 | Lower Panel Installation - Removable, Inboard Fixed Leading Edge |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 116A2100 | Lower Panel Installation, Inboard Fixed Leading Edge, Wing |

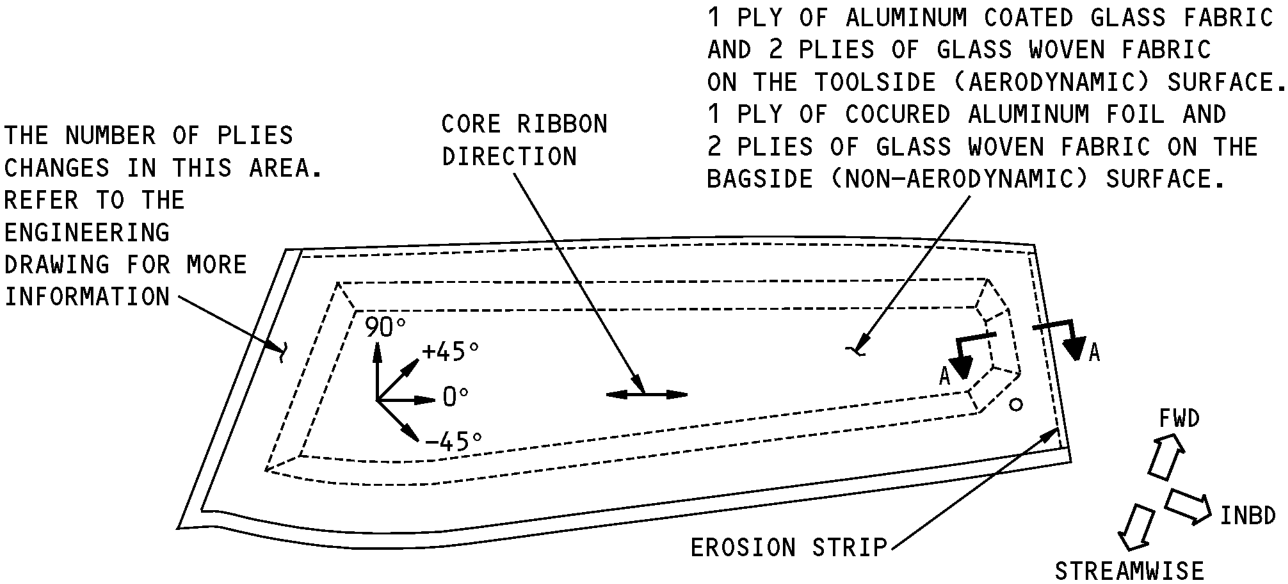



**Wing Inboard Leading Edge Lower Skin Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Skin, Closeout | 0.063 (1.6) | 2024-T3 clad sheet as given in QQ-A-250/5 | |
| [2] | Panel Bond Assembly Lower Removable | | | |
| | Skin Assembly | | Refer to Figure 3 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |

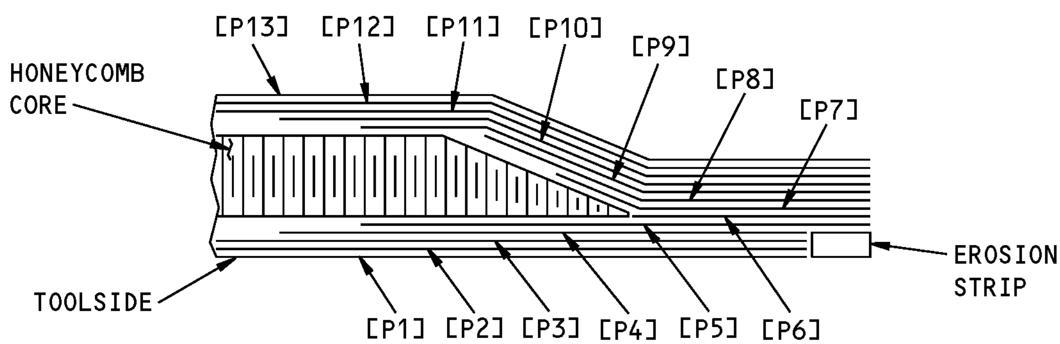
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION AND CORE RIBBON DIRECTION

A




A-A

NOTES

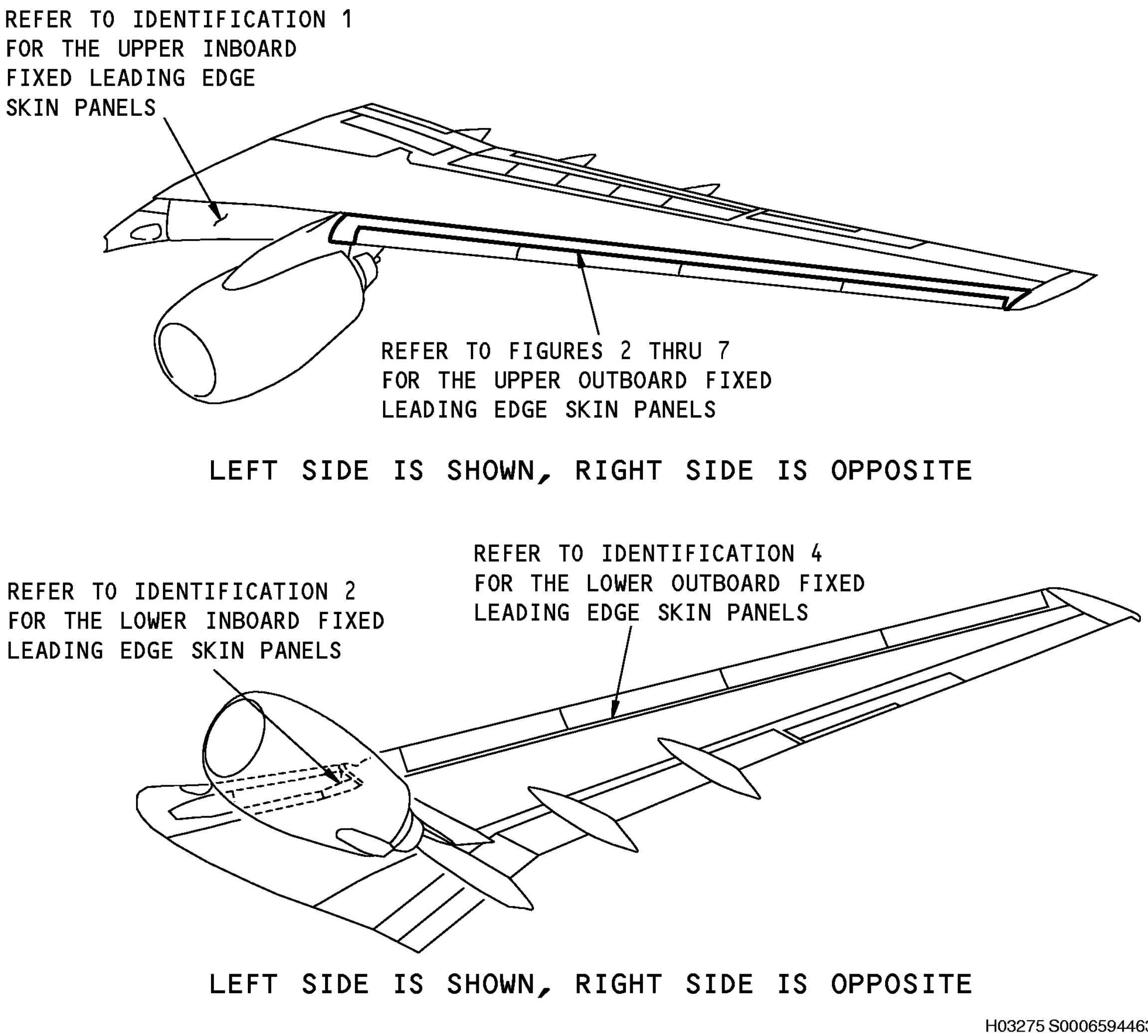
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Upper Removable Access Panel, Figure 2, Item [2]**
**Figure 3**

**Table 3**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [2] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A) |
| P2 to P5, and P9 to P12 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6 to P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P13 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B |

## IDENTIFICATION 3 - WING OUTBOARD FIXED LEADING EDGE UPPER SKIN PANELS

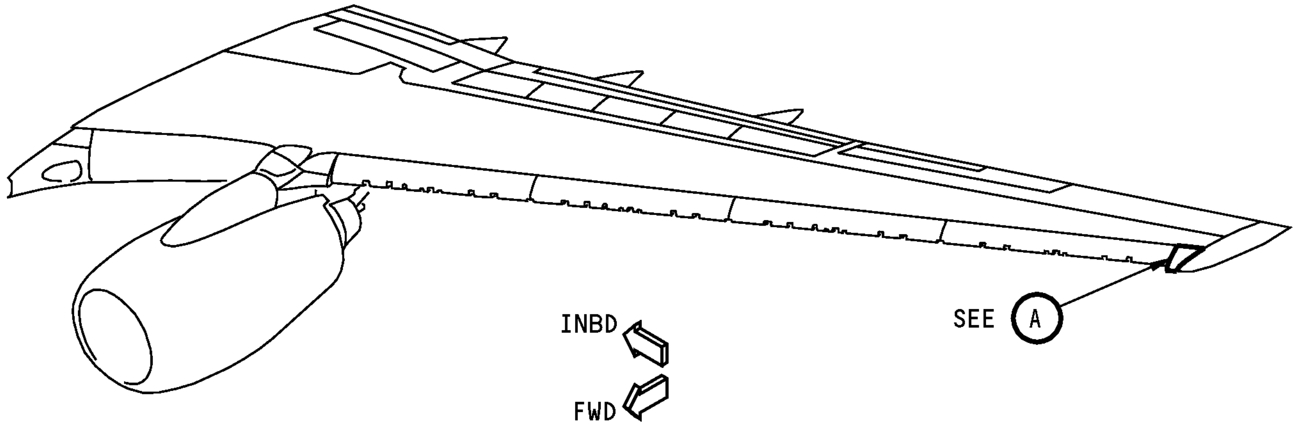


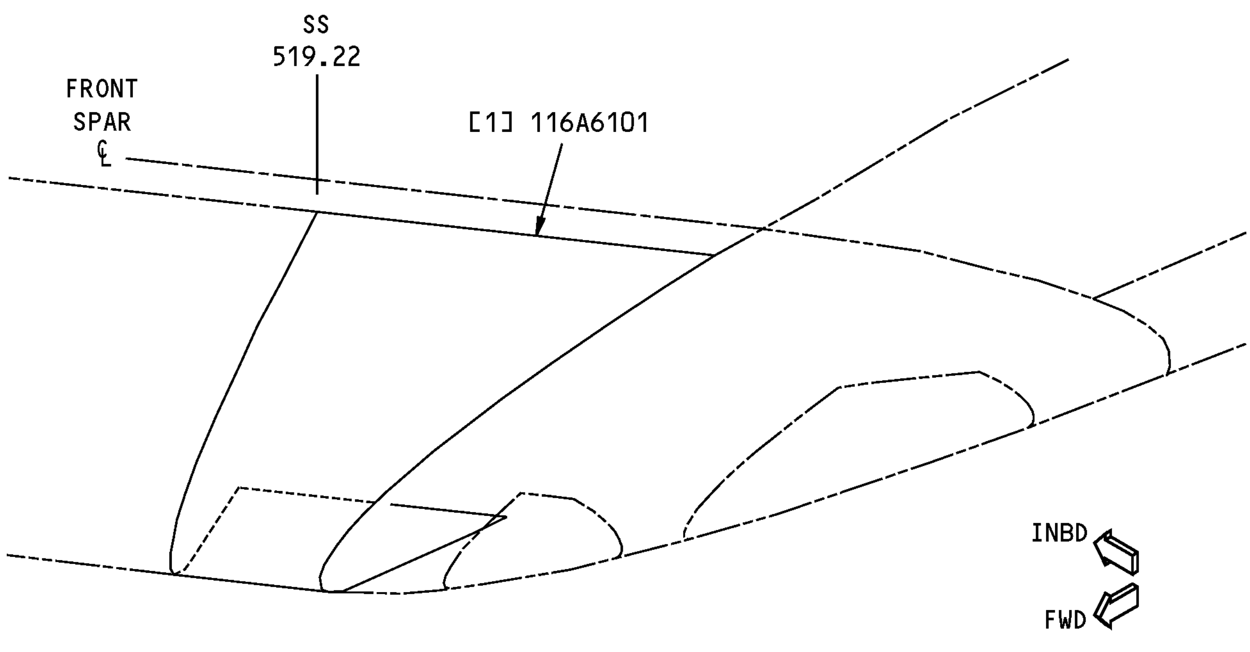

**Wing Outboard Fixed Leading Edge Upper Skin Panel Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A1001 | Wing Left - Product Collector |
| 116A0013 | Collector - Leading Edge, Wing Build-Up, Left Wing |
| 116A0015 | Functional Product Collector - Front Spar Leading Edge Join, Left Wing |
| 116A5410 | Installation - Upper Panel, Slat 1, Outboard Fixed Leading Edge |
| 116A5420 | Installation - Upper Panel, Slat 2, Outboard Fixed Leading Edge |
| 116A5430 | Installation - Upper Panel, Slat 3, Outboard Fixed Leading Edge |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 116A5440 | Installation - Upper Panel, Slat 4, Outboard Fixed Leading Edge |
| 116A6100 | Leading Edge Installation, SS 519.22 to WBL 658.17, Outboard Fixed Leading Edge |
| 116A6200 | Gap Cover Installation - Outboard Fixed Leading Edge |
| 116A6201 | Upper Panel Assembly - Gap Cover, Outboard Fixed Leading Edge |




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

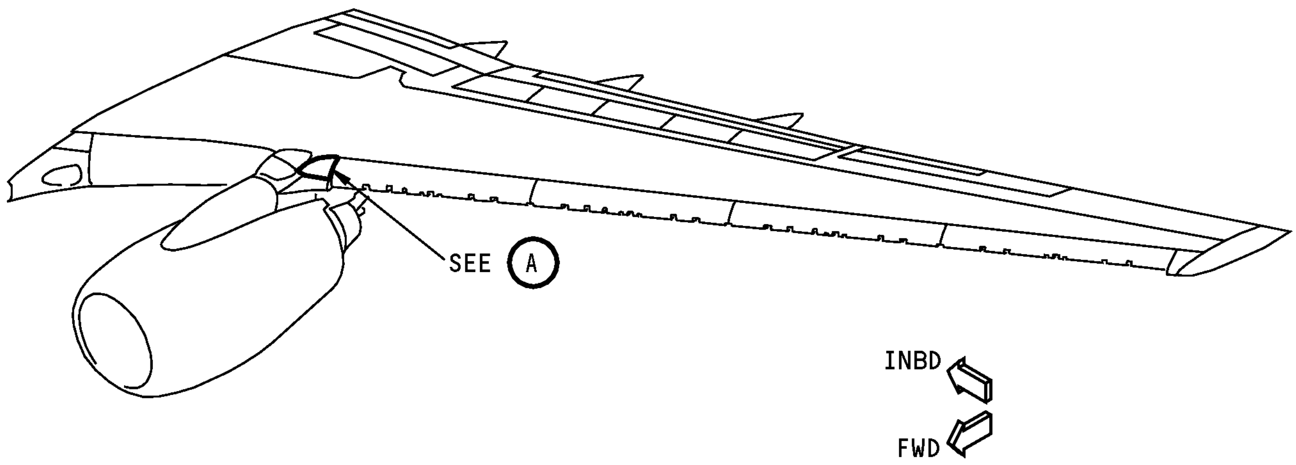
A

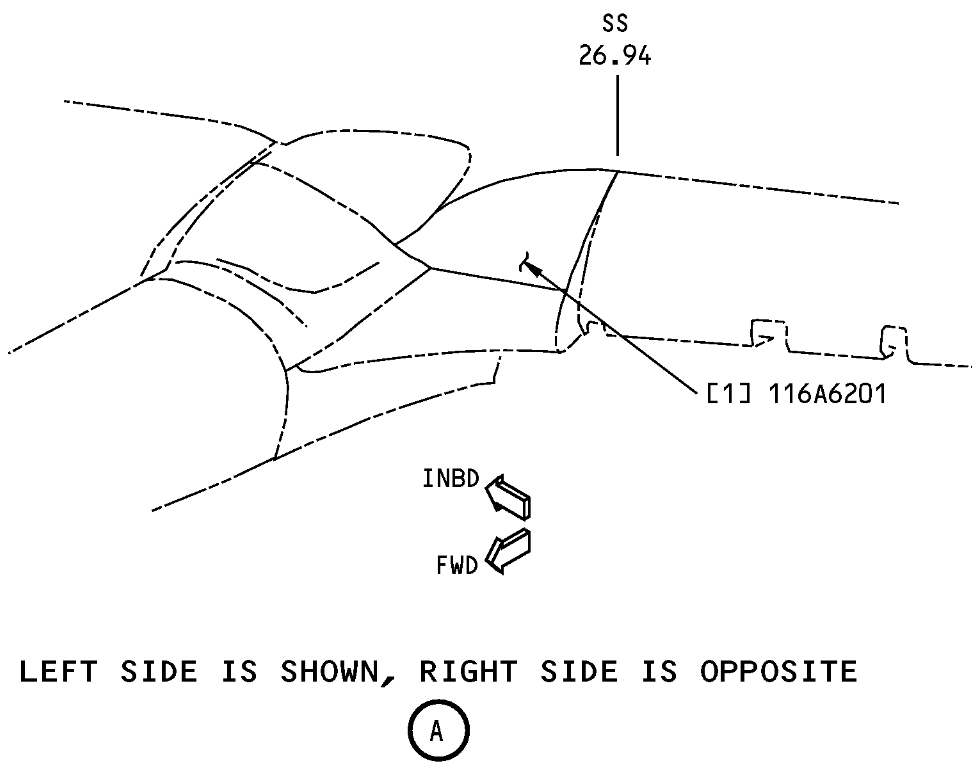
**Wing Outboard Fixed Leading Edge Upper Skin Panel Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Upper Panel SS 519.22 to WBL 658.17 | 0.063 (1.6) | 2024-T42 clad sheet as given in QQ-A-250/5 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



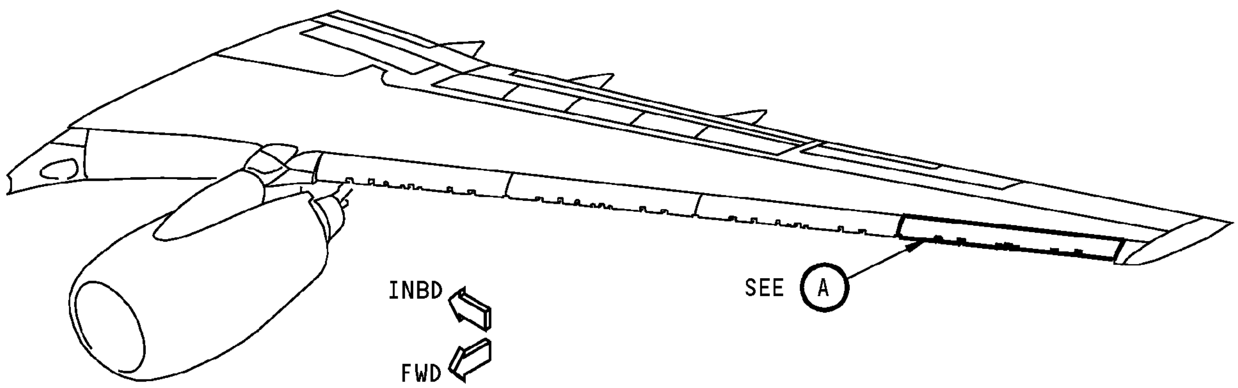



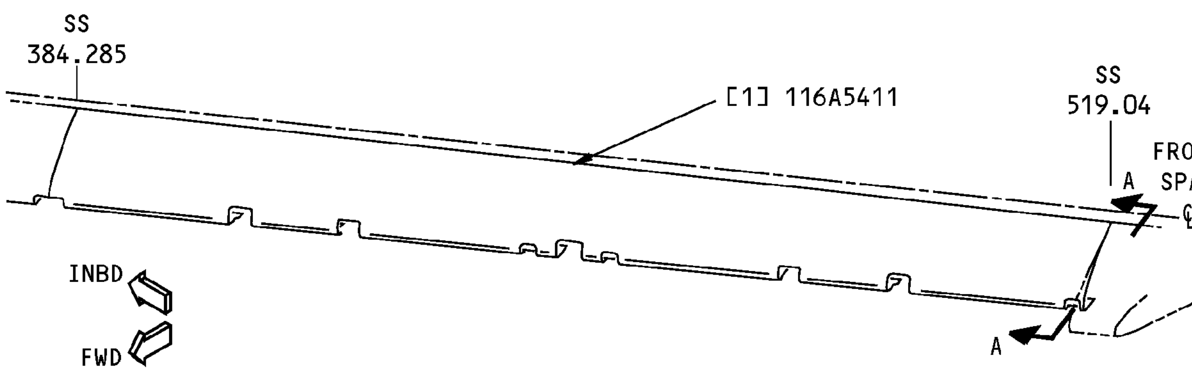

**Wing Outboard Fixed Leading Edge Gap Cover Identification**
**Figure 3**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Upper Panel, Gap Cover | 0.080 (2.03) | 2024-T42 clad sheet as given in QQ-A-250/5 | |

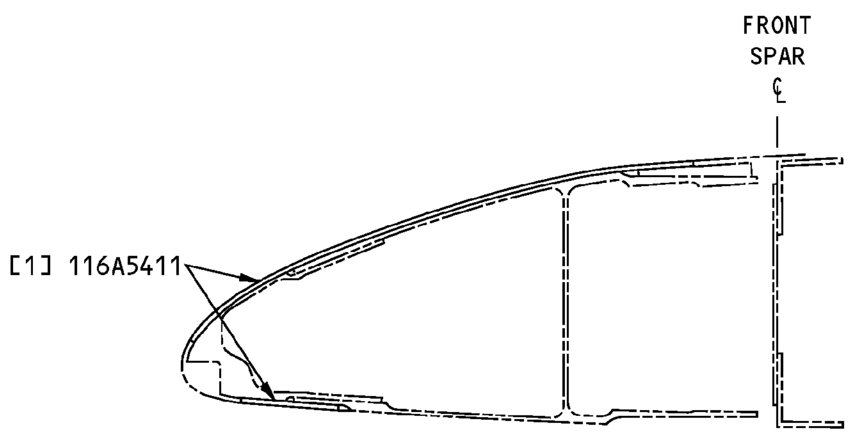
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A



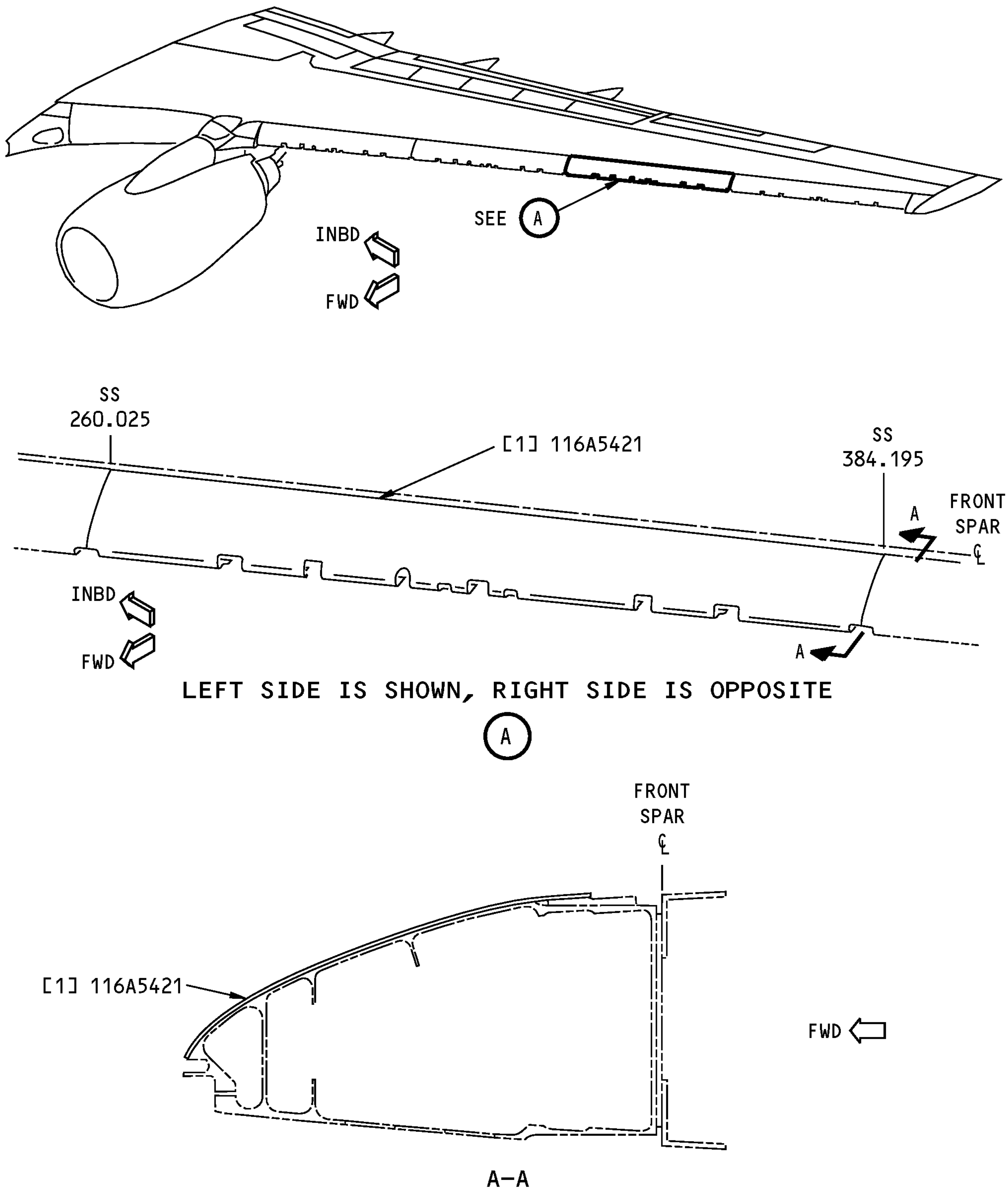

A-A

**Wing Outboard Fixed Leading Edge Slat No.1 Upper Panel Identification**
**Figure 4**

Table 4:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Bond Panel Assembly Slat 1 | | | |
| | Skin Assembly | | See Figure 8 | |
| | Core (7) | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

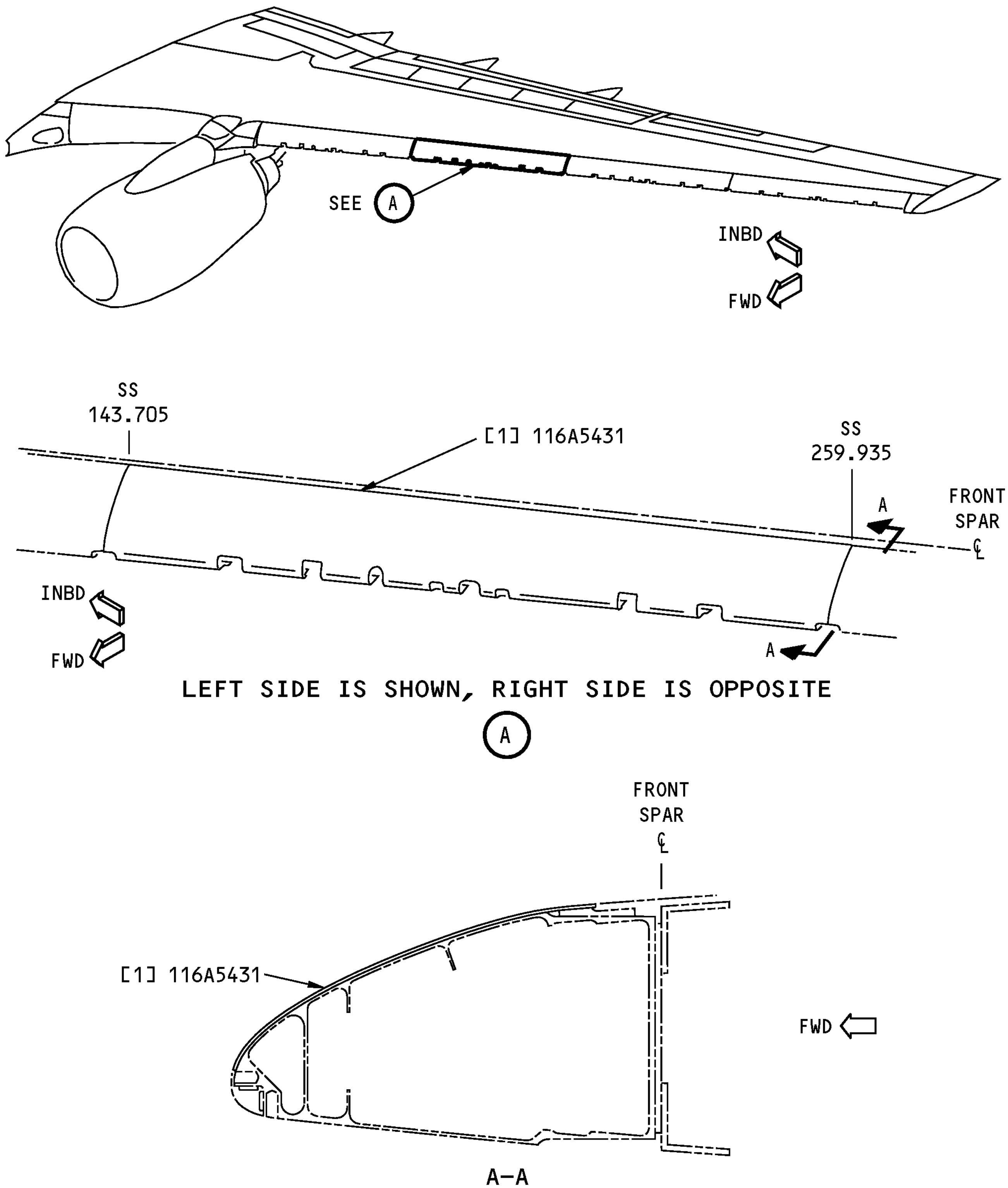



**Wing Outboard Fixed Leading Edge Slat No.2 Upper Panel Identification**
**Figure 5**

**Table 5**:

| LIST OF MATERIALS FOR FIGURE 5 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Bond Panel Assembly Slat 2 | | | |
| | Skin Assembly | | See Figure 9 | |
| | Core (7) | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

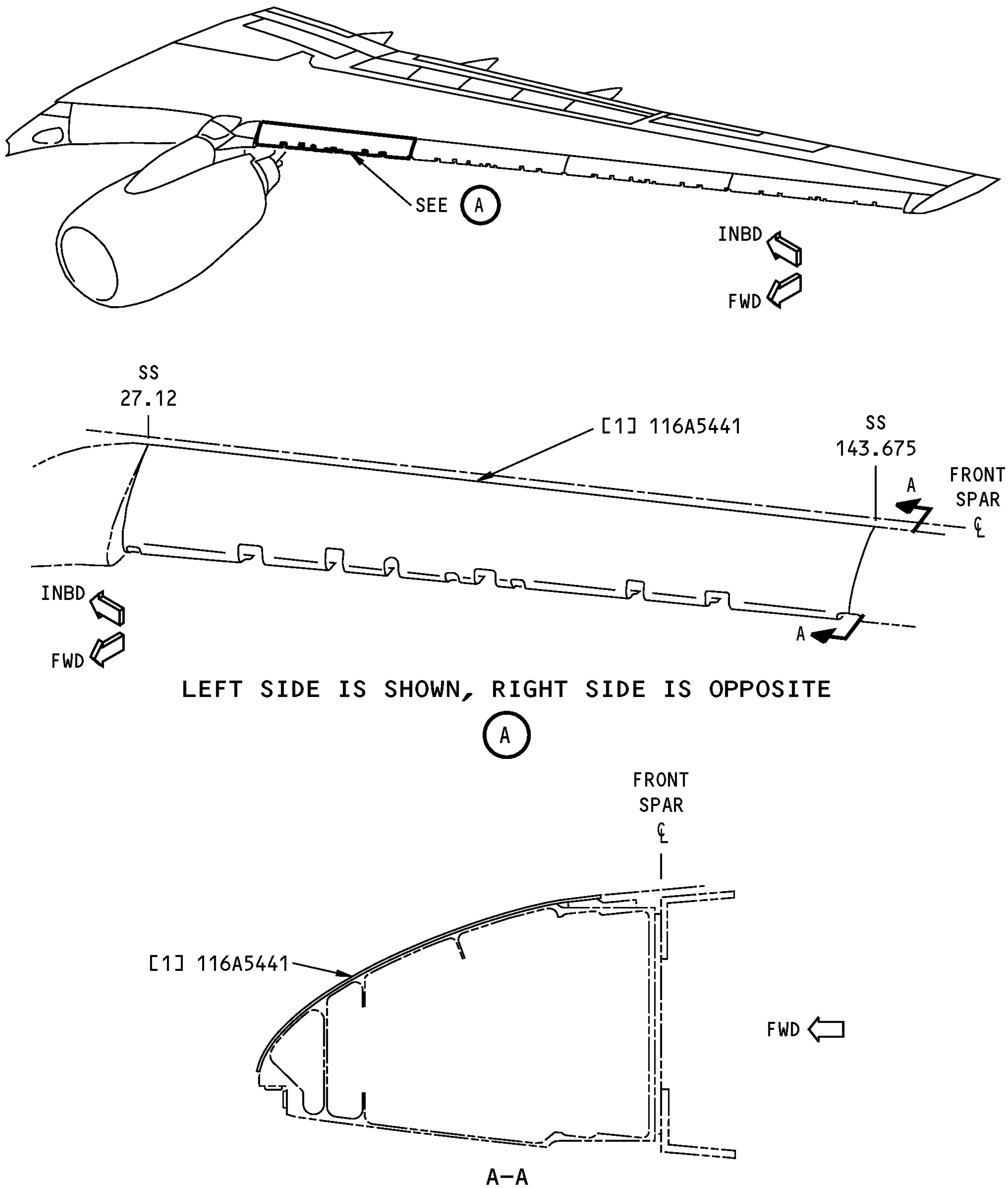



**Wing Outboard Fixed Leading Edge Slat No.3 Upper Panel Identification**
**Figure 6**

**Table 6**:

| LIST OF MATERIALS FOR FIGURE 6 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Bond Panel Assembly Slat 3 | | | |
| | Skin Assembly | | See Figure 10 | |
| | Core (7) | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

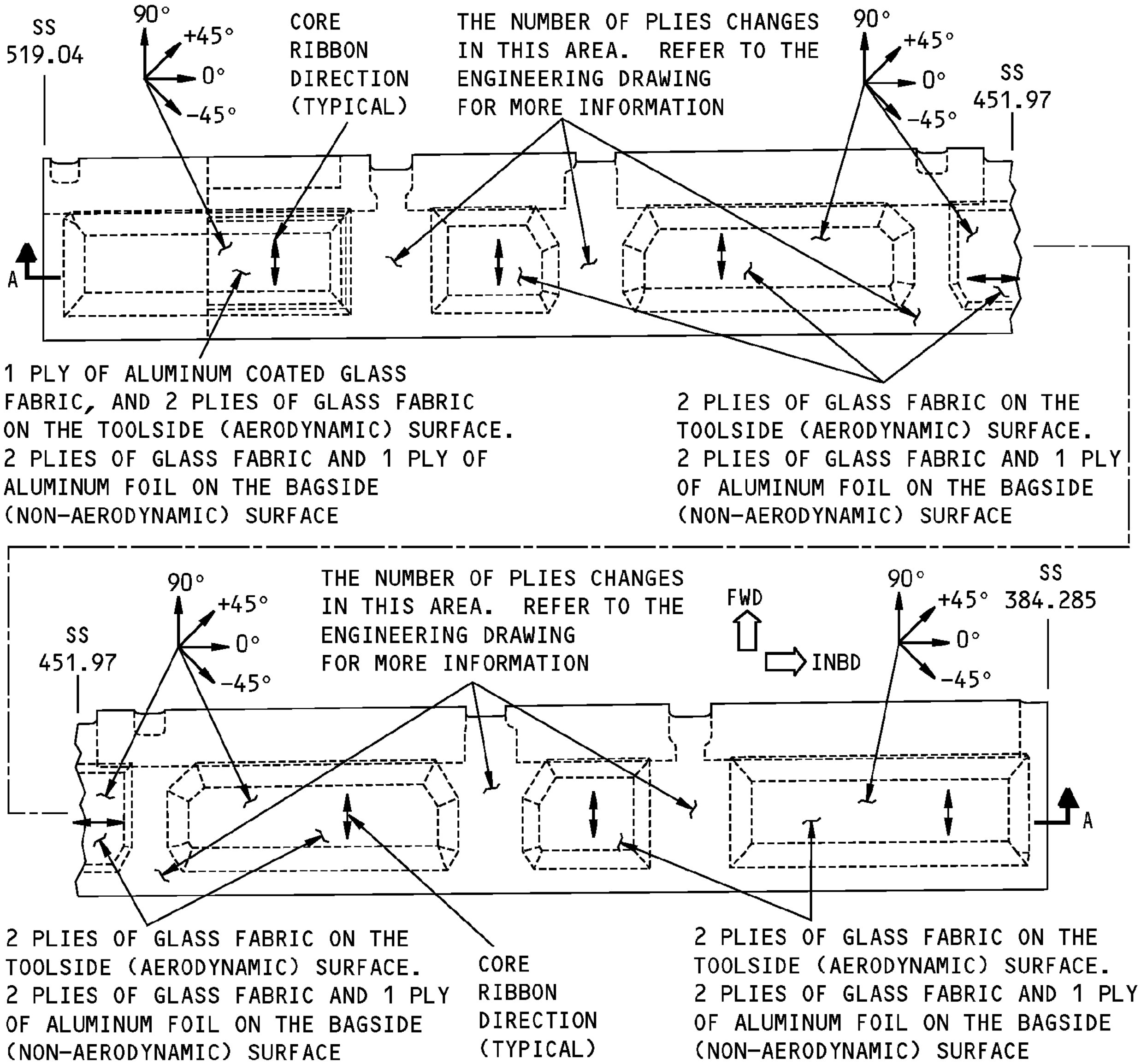



**Wing Outboard Fixed Leading Edge Slat No.4 Upper Panel Identification**
**Figure 7**

**Table 7**:

| LIST OF MATERIALS FOR FIGURE 7 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Bond Panel Assembly Slat 4 | | | |
| | Skin Assembly | | See Figure 11 | |
| | Core (7) | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



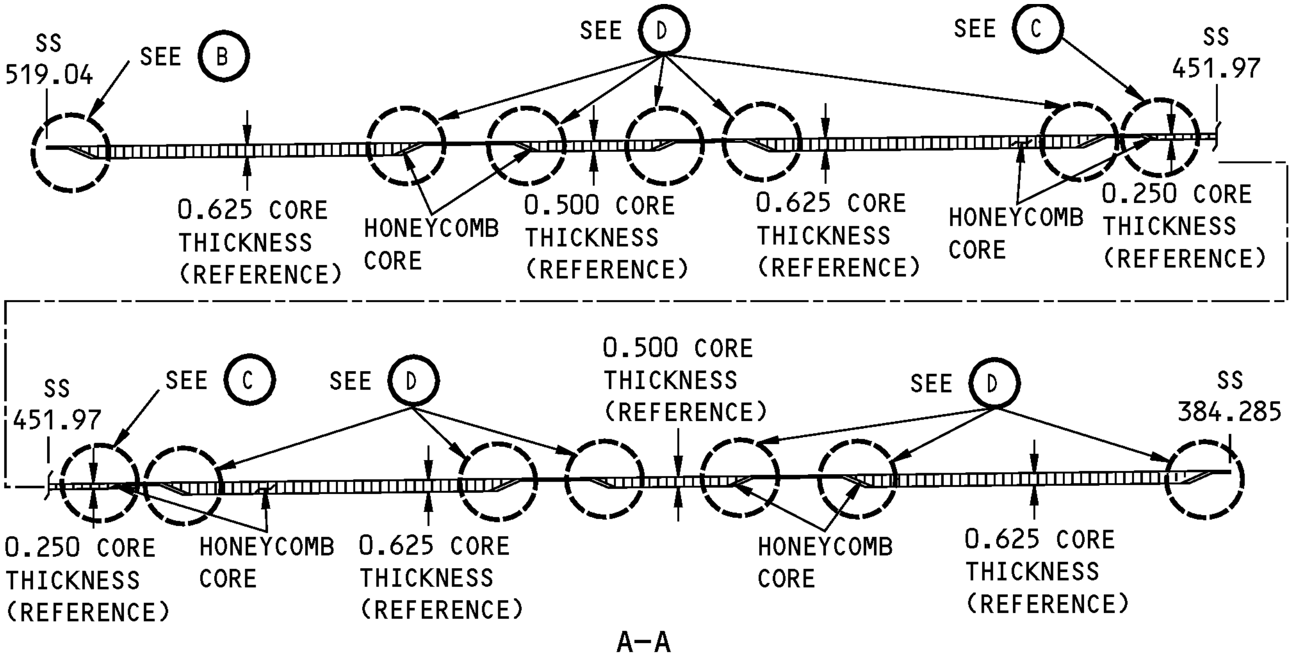

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

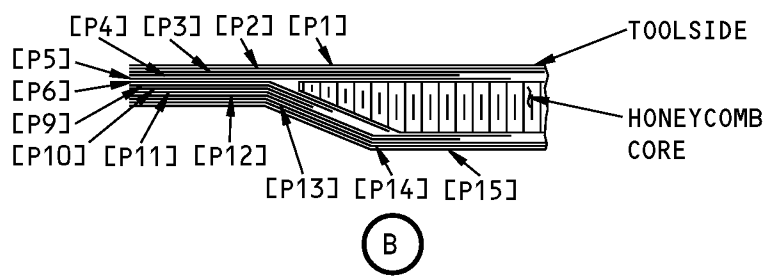
A

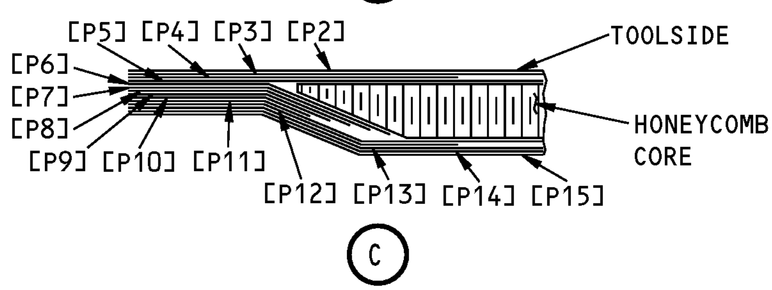
NOTES

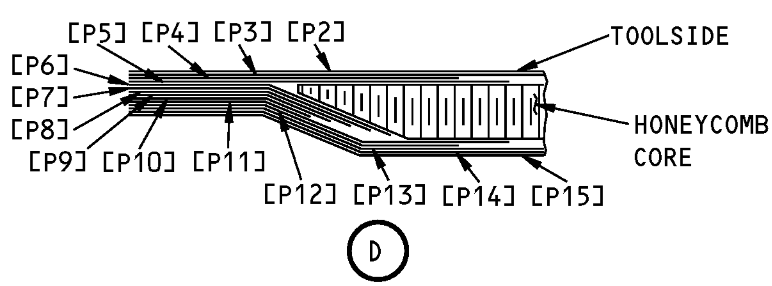
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.1 Panel, Figure 4, Item [1]**
**Figure 8 (Sheet 1 of 2)**



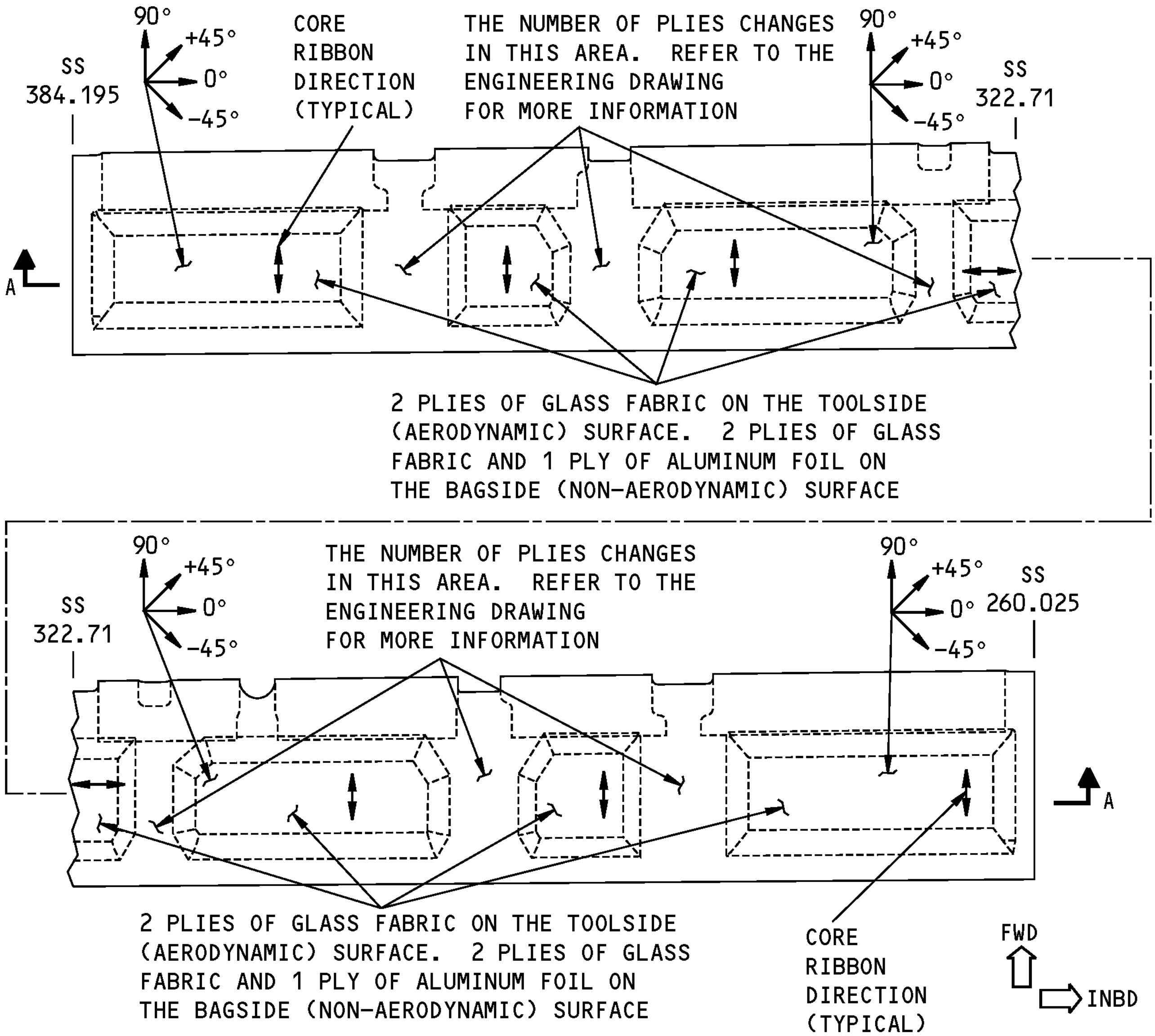




**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.1 Panel, Figure 4, Item [1]**
**Figure 8 (Sheet 2 of 2)**

**Table 8**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 8 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A) |
| P2 to P5, and P11 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, P9, and P10 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | —— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material |



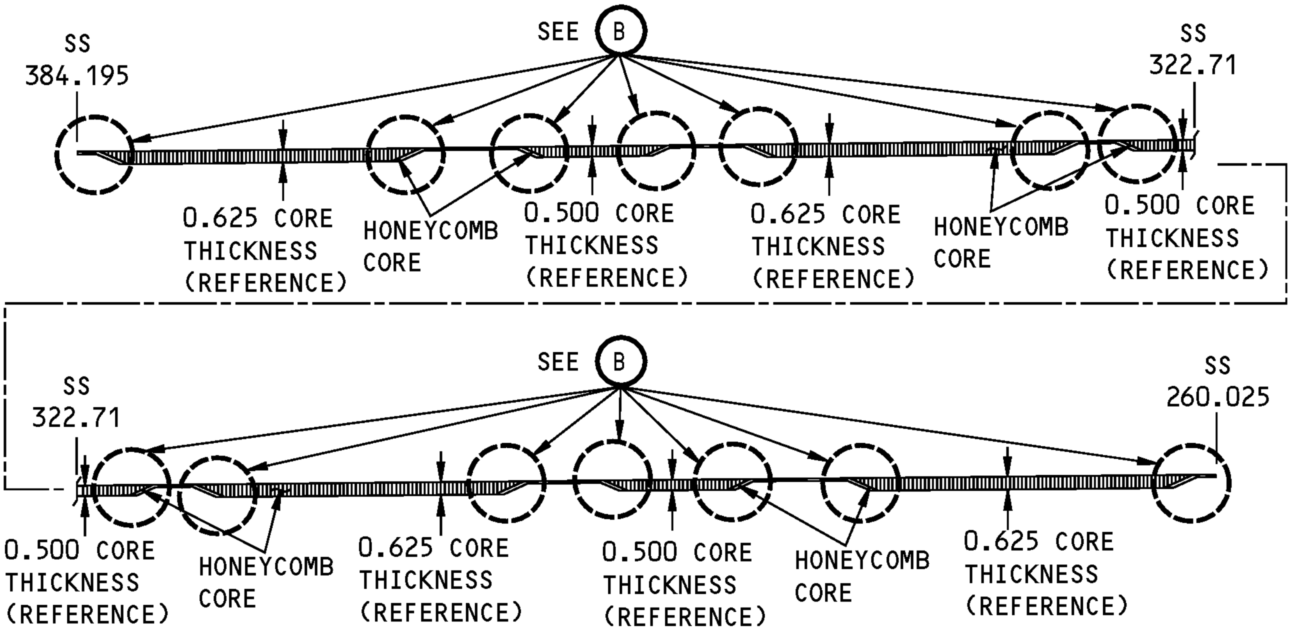

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

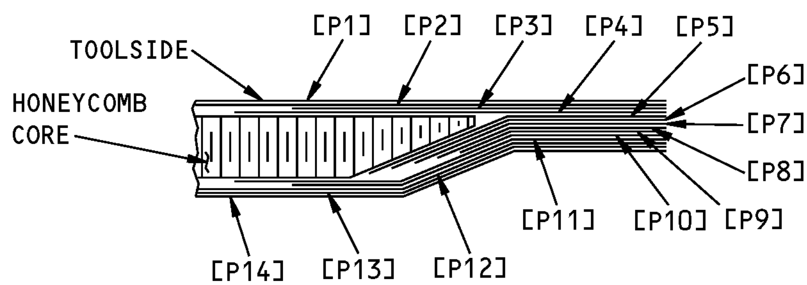
NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.2 Panel, Figure 5, Item [1]**
**Figure 9 (Sheet 1 of 2)**




A-A

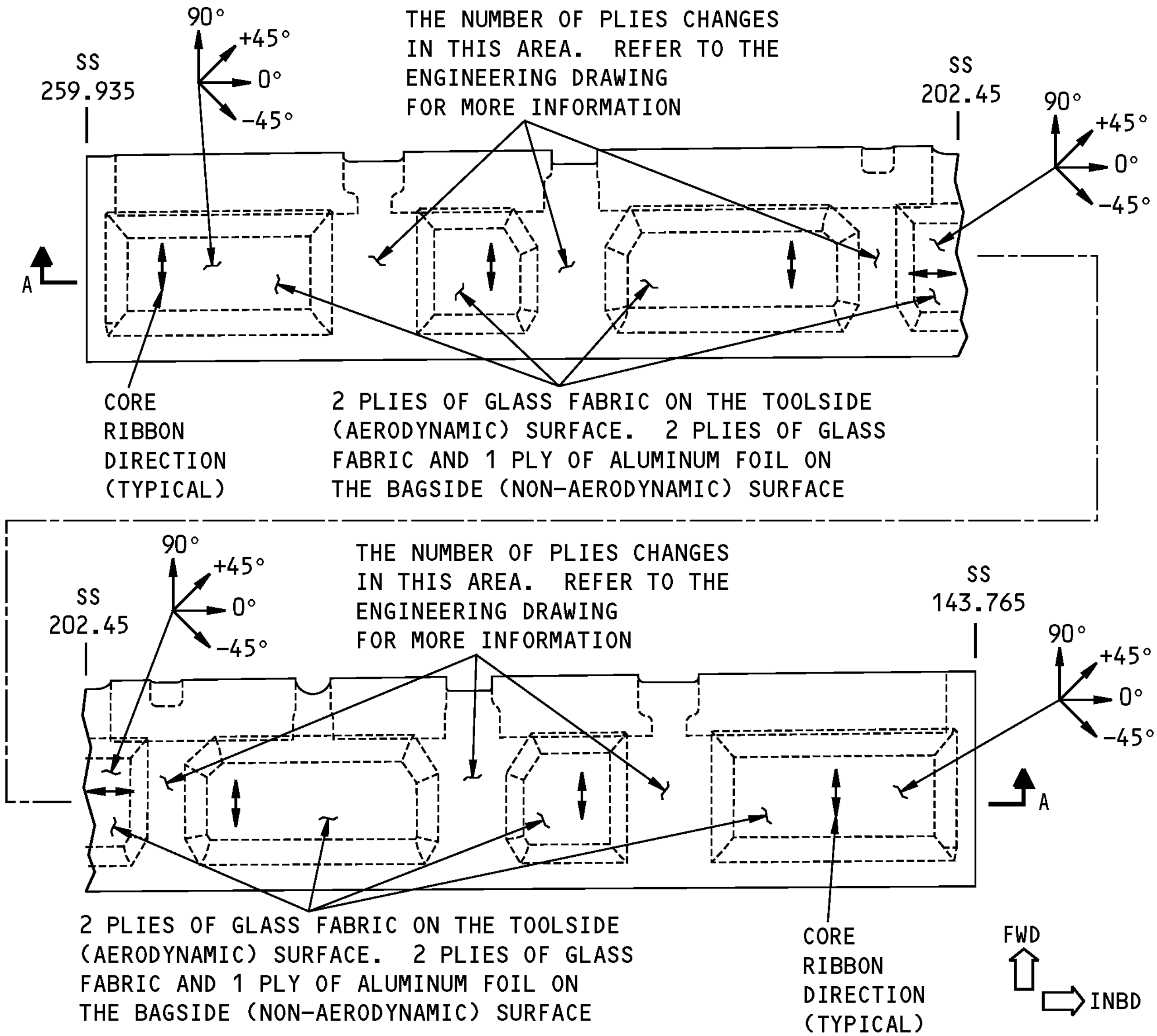



B

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.2 Panel, Figure 5, Item [1]**
**Figure 9 (Sheet 2 of 2)**

**Table 9**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 9 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 to P4, and P10 to P13 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P5, P6, P8, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P7 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P14 | —— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material |



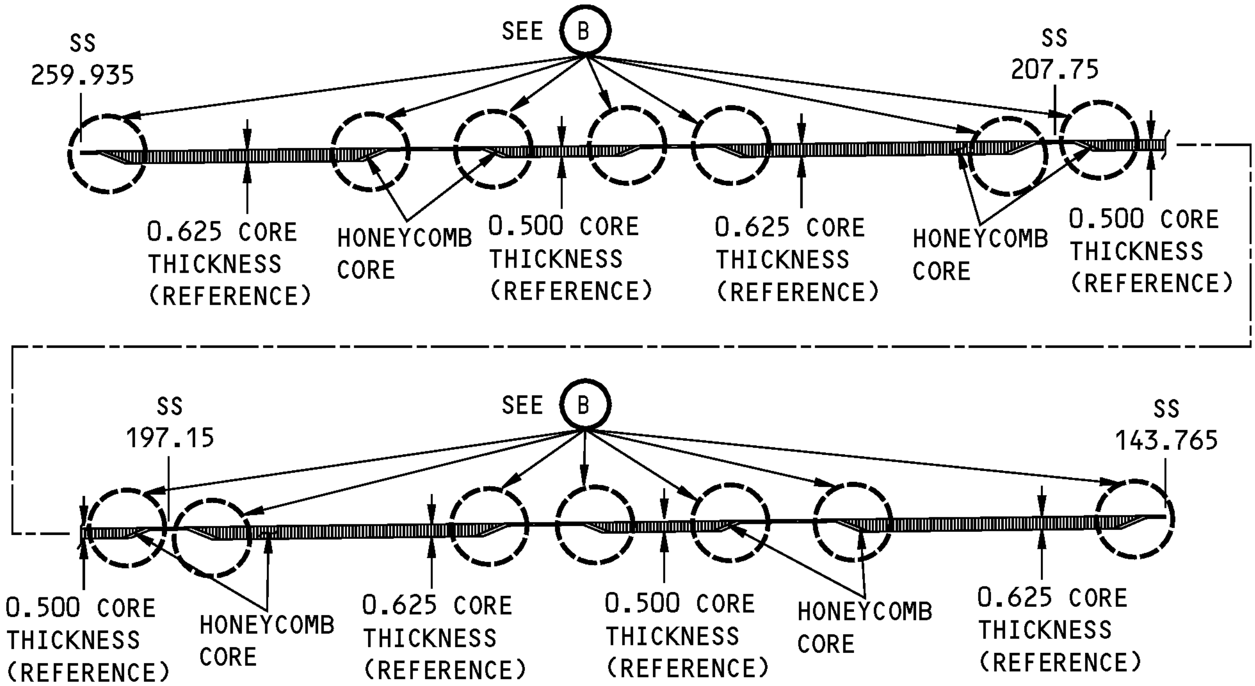

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

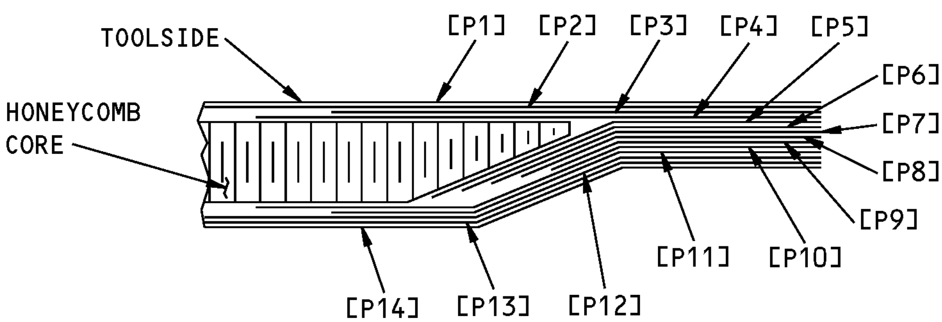
NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.3 Panel, Figure 6, Item [1]**
**Figure 10 (Sheet 1 of 2)**




A-A

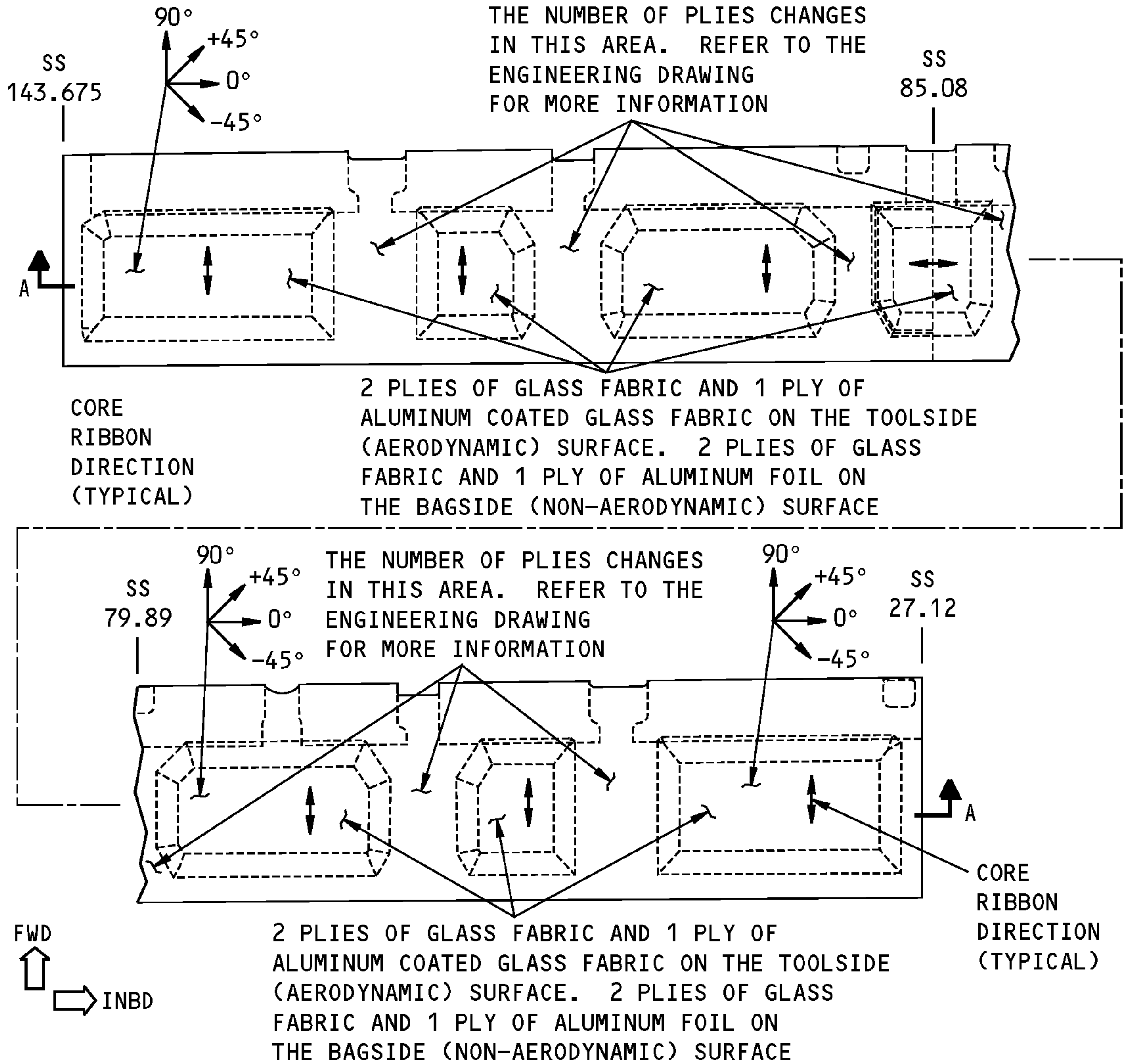



B

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.3 Panel, Figure 6, Item [1]
Figure 10 (Sheet 2 of 2)

Table 10:

| PLY MATERIAL AND DIRECTION FOR FIGURE 10 | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1 to P4, and P10 to P13 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P5, P6, P8, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P7 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P14 | —— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material |



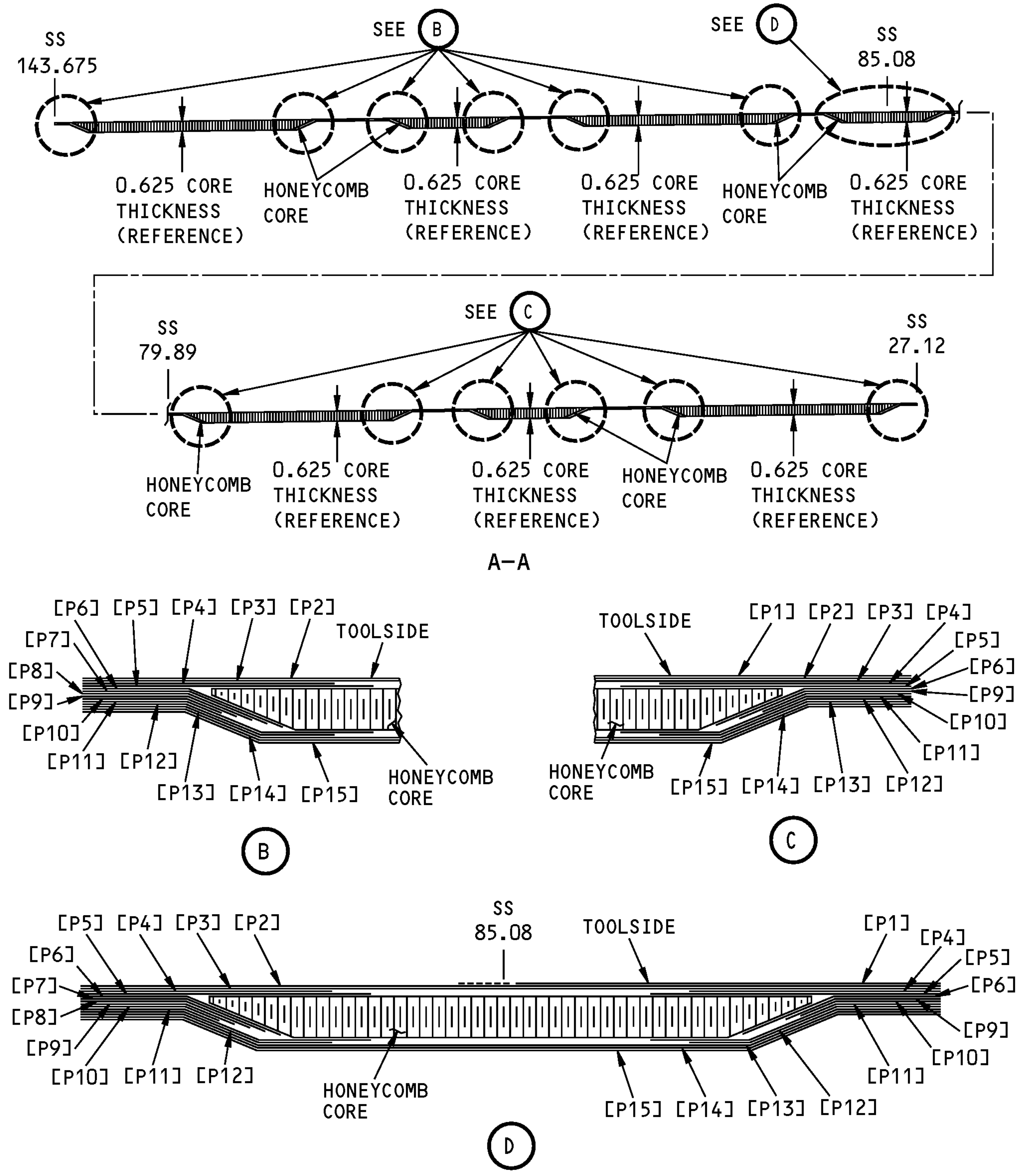

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 11 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.4 Panel, Figure 7, Item [1]**
**Figure 11 (Sheet 1 of 2)**

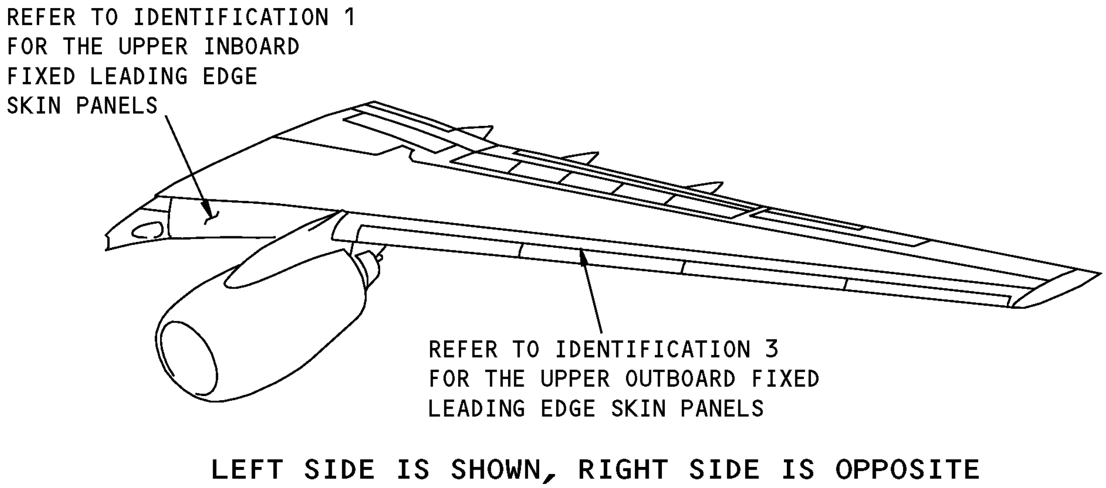


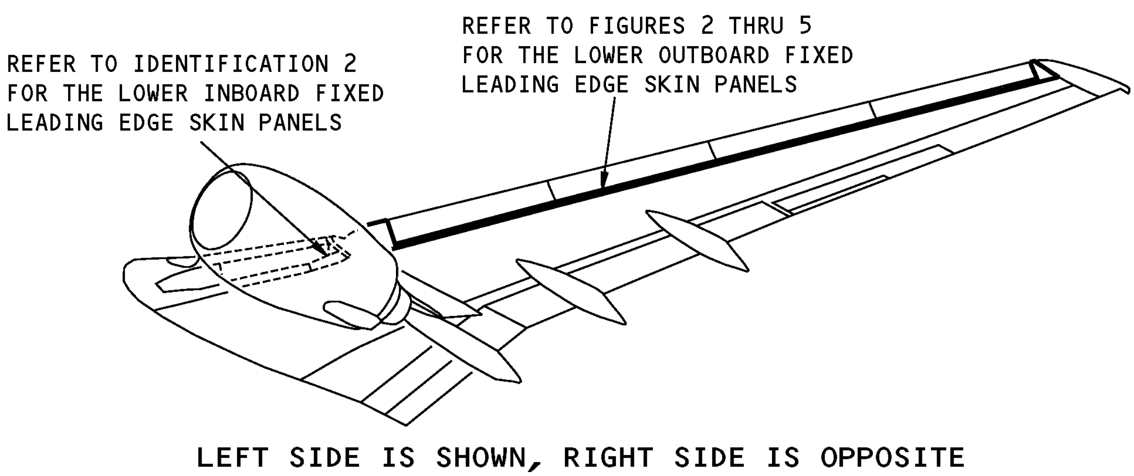

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.4 Panel, Figure 7, Item [1]**
**Figure 11 (Sheet 2 of 2)**

**Table 11**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 11 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A) |
| P2 to P5, and P11 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, P9, and P10 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | —— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material |

## IDENTIFICATION 4 - WING OUTBOARD FIXED LEADING EDGE LOWER SKIN PANELS



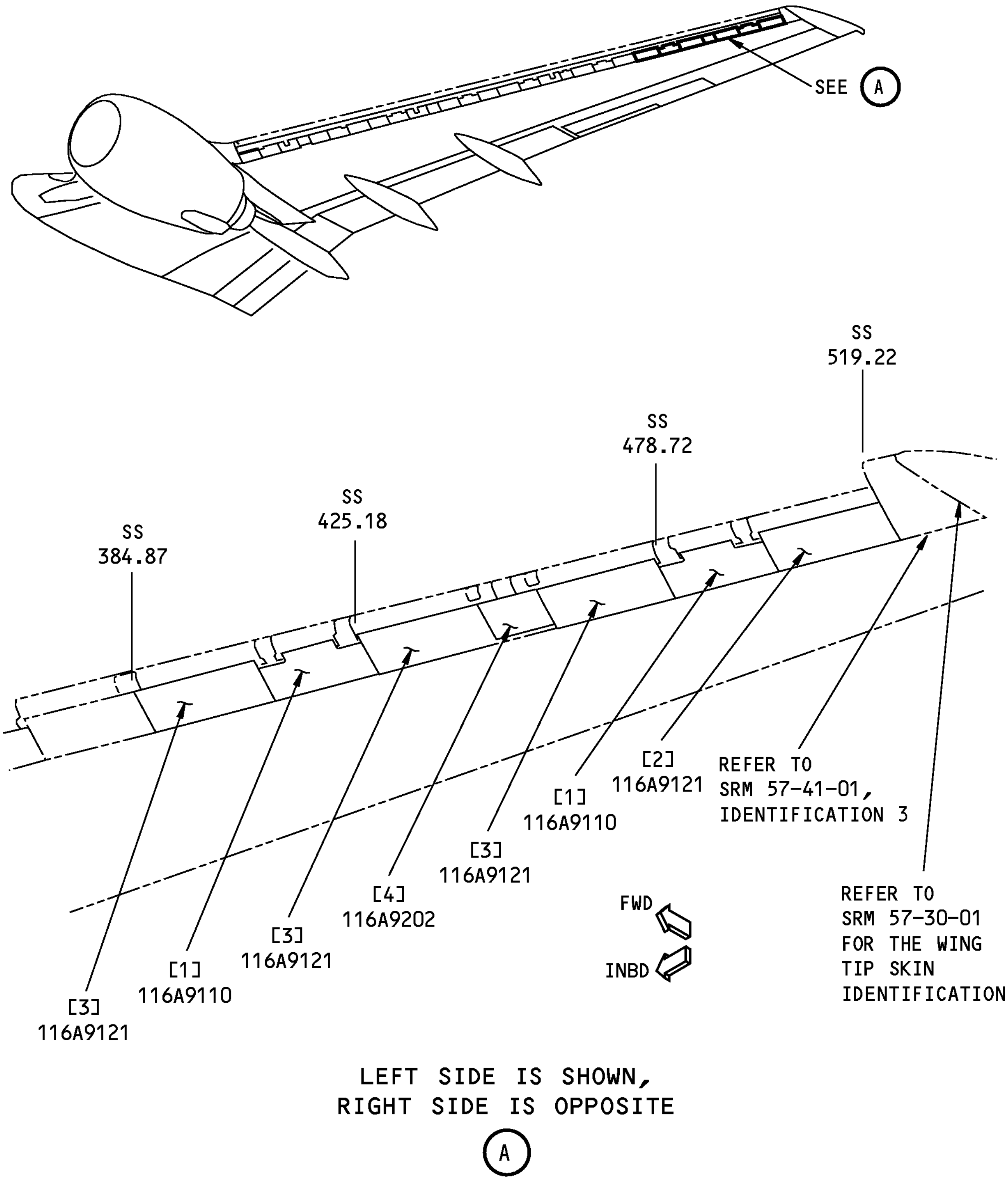




**Wing Outboard Fixed Leading Edge Lower Skin Panel Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A0101 | Final Assembly - Product Collector |
| 001A1001 | Wing Left - Product Collector |
| 116A0010 | Fixed Leading Edge Collector - Final Assembly |
| 116A0011 | Fixed Leading Edge Collector - Post Paint - Left Wing |
| 116A8500 | Access Door Installation - Outboard Fixed Leading Edge |
| 116A8510 | Door Assembly - Engine Fuel Shutoff/Defuel Valves, Outboard Fixed Leading Edge |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 116A9100 | Removable Lower Panel Installation - Outboard Fixed Leading Edge |
| 116A9110 | Panel Assembly - Lower, Type I, Outboard Fixed Leading Edge |
| 116A9200 | Seal Door Installation - Actuator Body, Outboard Fixed Leading Edge |
| 116A9202 | Assembly - Actuator Door, Outboard Fixed Leading Edge |

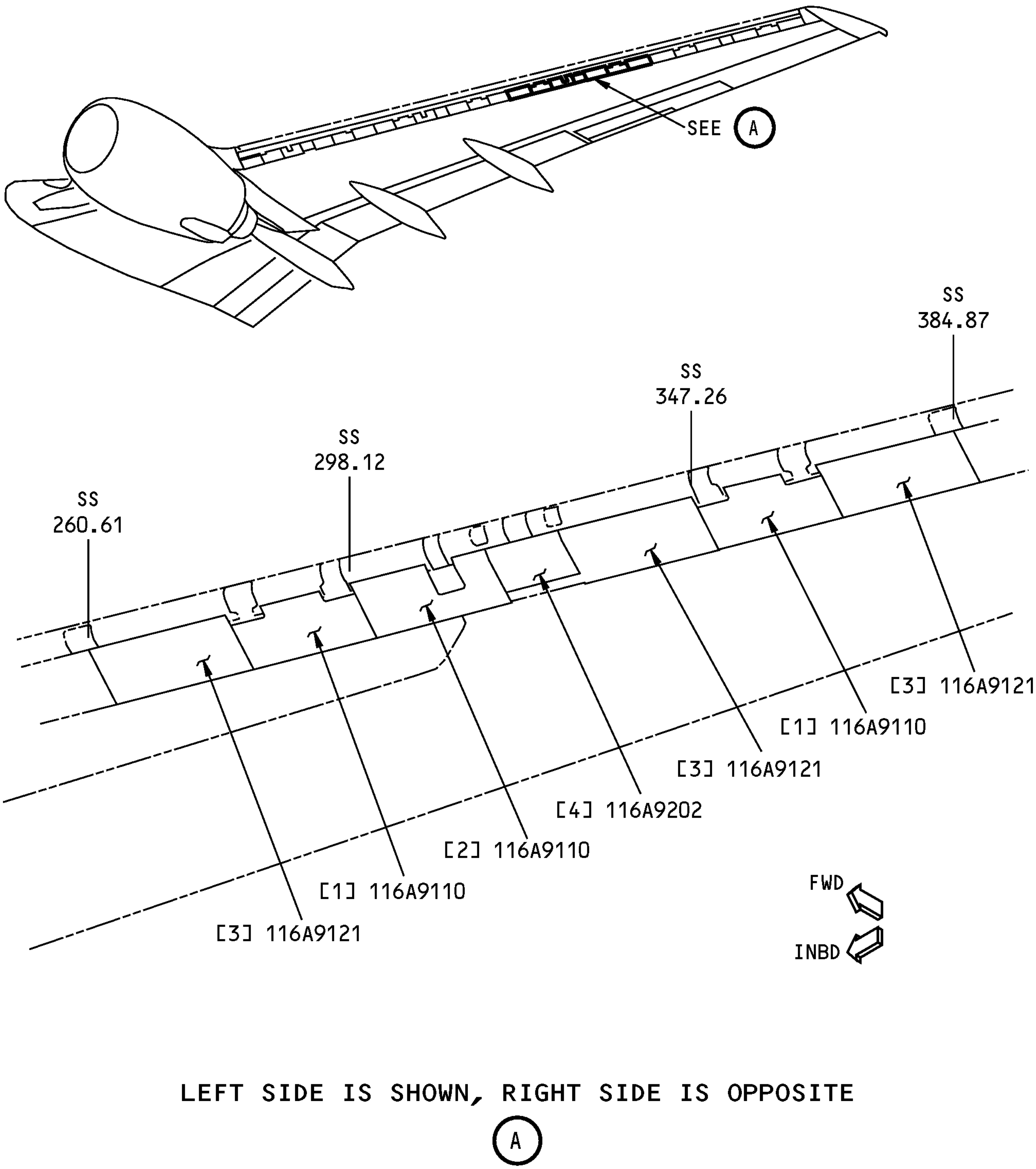



**Fixed Leading Edge Lower Outboard Removable Panel Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 SS 384.87 TO SS 519.22 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Panel Bond Assembly Lower Removable (2) | | | |
| | Skin Assembly | | Refer to Figure 6 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [2] | Panel Bond Assembly Lower Removable | | | |
| | Skin Assembly | | Refer to Figure 7 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [3] | Panel Bond Assembly Lower Removable (3) | | | |
| | Skin Assembly | | Refer to Figure 8 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [4] | Door | 0.100 (2.54) | 7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 9 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

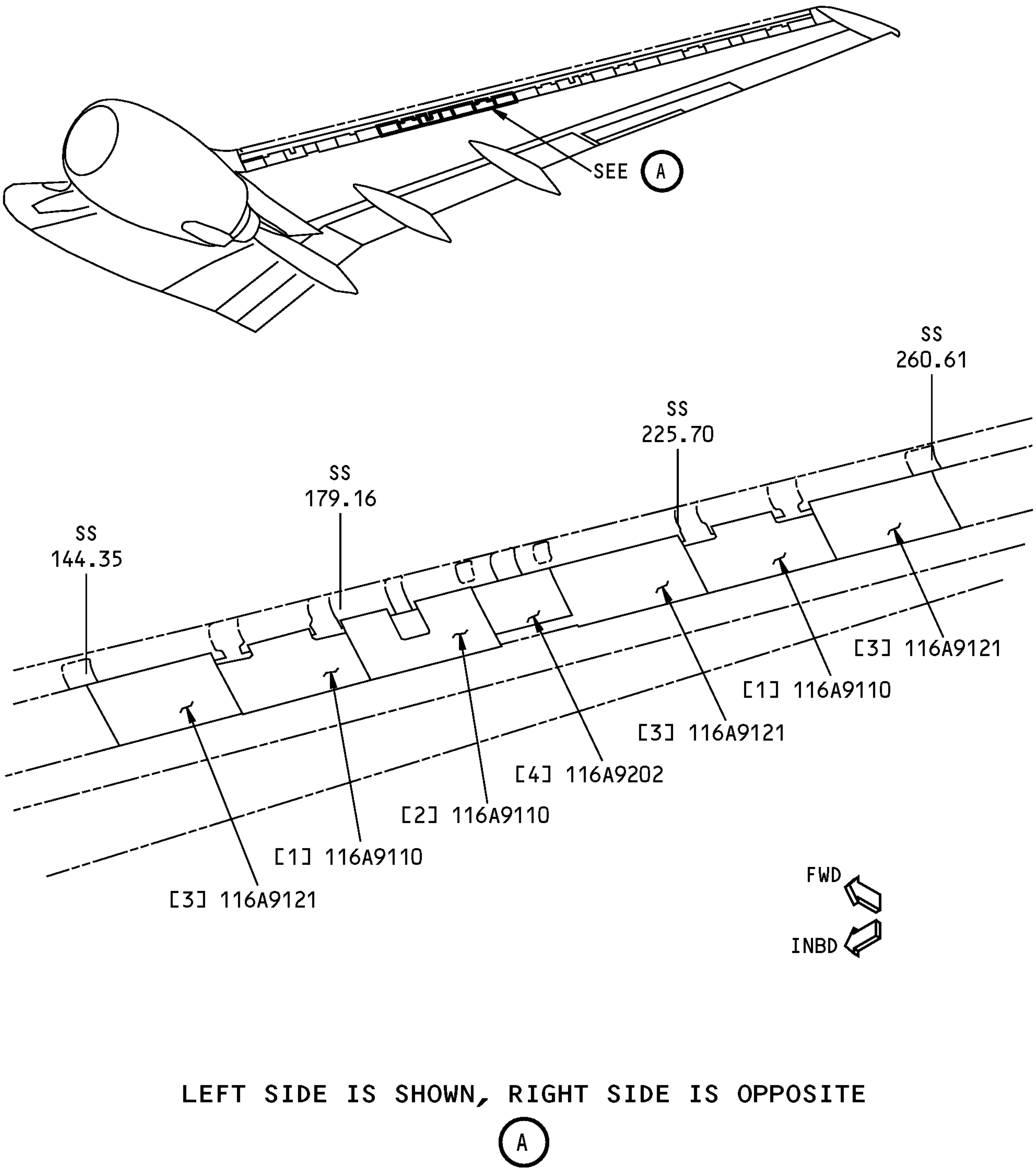



**Fixed Leading Edge Lower Outboard Removable Panel Identification**
**Figure 3**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 SS 260.61 TO SS 384.87 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Panel Bond Assembly Lower Removable (2) | | | |
| | Skin Assembly | | Refer to Figure 10 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [2] | Panel Bond Assembly Lower Removable | | | |
| | Skin Assembly | | Refer to Figure 11 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [3] | Panel Bond Assembly Lower Removable (3) | | | |
| | Skin Assembly | | Refer to Figure 12 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [4] | Door | 0.100 (2.54) | 7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 13 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

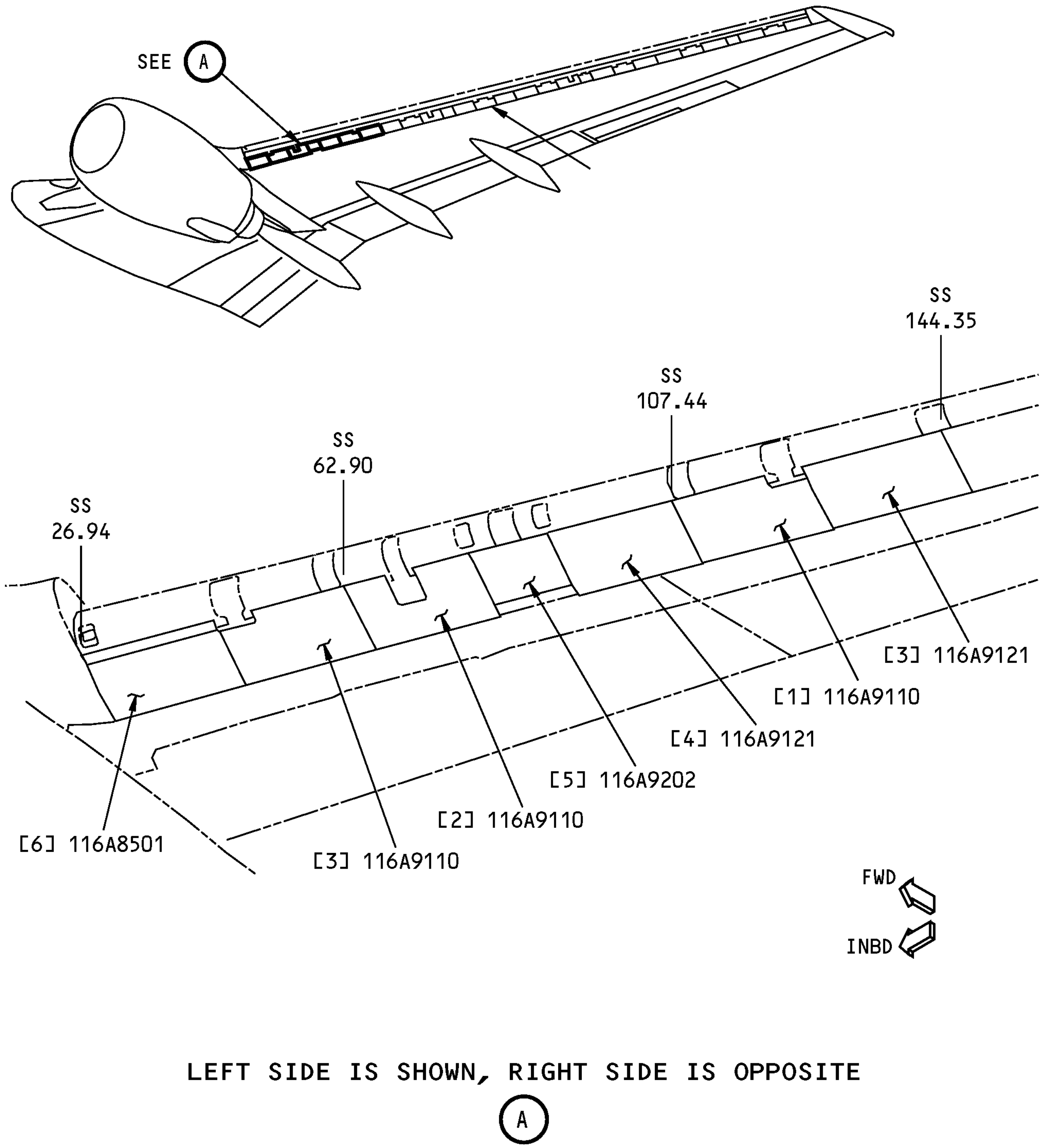



**Fixed Leading Edge Lower Outboard Removable Panel Identification**
**Figure 4**

**Table 4**:

| LIST OF MATERIALS FOR FIGURE 4 SS 144.35 TO SS 260.61 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Panel Bond Assembly Lower Removable (2) | | | |
| | Skin Assembly | | Refer to Figure 14 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [2] | Panel Bond Assembly Lower Removable | | | |
| | Skin Assembly | | Refer to Figure 15 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [3] | Panel Bond Assembly Lower Removable (3) | | | |
| | Skin Assembly | | Refer to Figure 16 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [4] | Door | 0.100 (2.54) | 7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 17 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

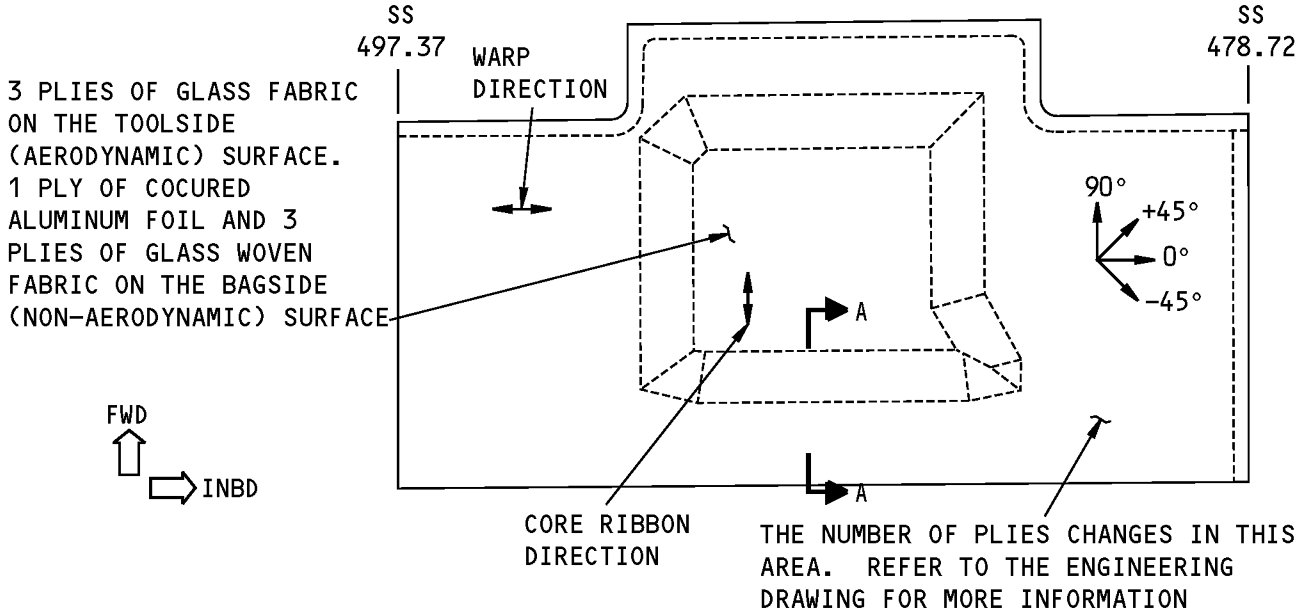



**Fixed Leading Edge Lower Outboard Removable Panel Identification**
**Figure 5**

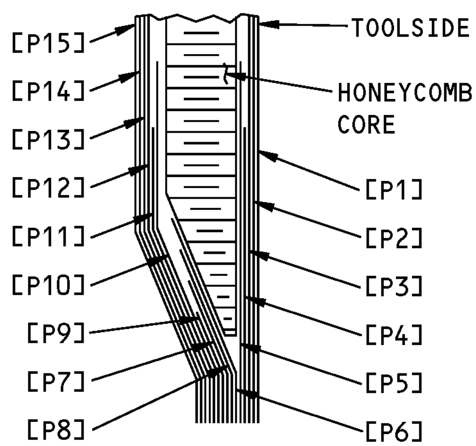
**Table 5**:

| LIST OF MATERIALS FOR FIGURE 5 SS 26.94 TO SS 144.35 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Panel Bond Assembly Lower Removable | | | |
| | Skin Assembly | | Refer to Figure 18 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [2] | Panel Bond Assembly Lower Removable | | | |
| | Skin Assembly | | Refer to Figure 19 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [3] | Panel Bond Assembly Lower Removable | | | |
| | Skin Assembly | | Refer to Figure 20 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [4] | Panel Bond Assembly Lower Removable (2) | | | |
| | Skin Assembly | | Refer to Figure 21 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [5] | Door | 0.100 (2.54) | 7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 22 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13) | |
| [6] | Access Door, Bonded Assembly | | | |
| | Skin Assembly | | Refer to Figure 23 | |
| | Core | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION
A




A-A

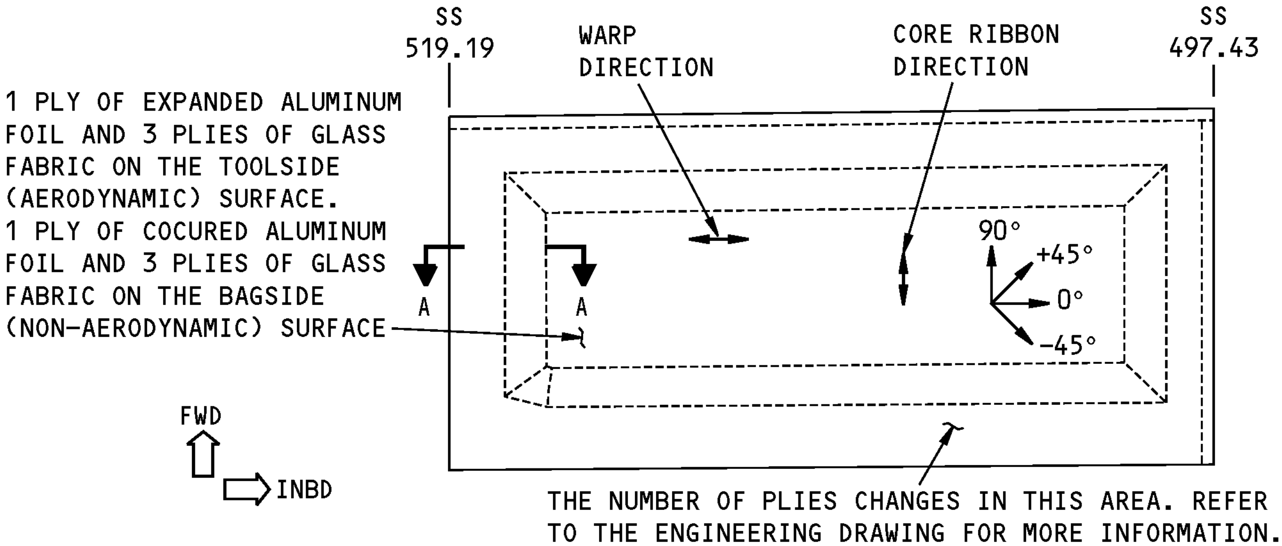
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO PRODUCTION DRAWINGS FOR SIMILAR PANELS IN THIS AREA.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 2, Item [1]**
**Figure 6**

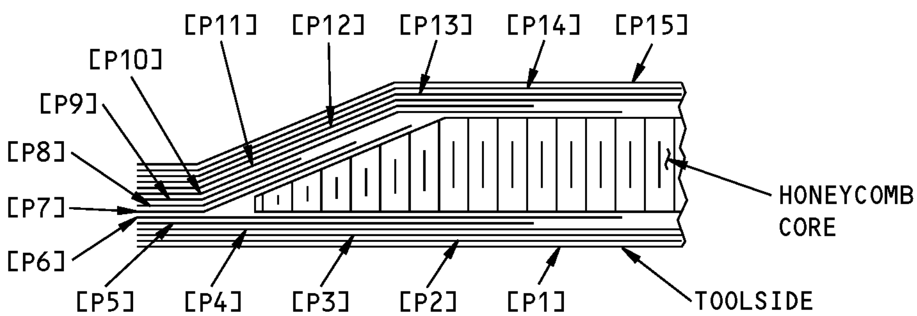
Table 6:

| PLY MATERIAL AND DIRECTION FOR FIGURE 6 | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

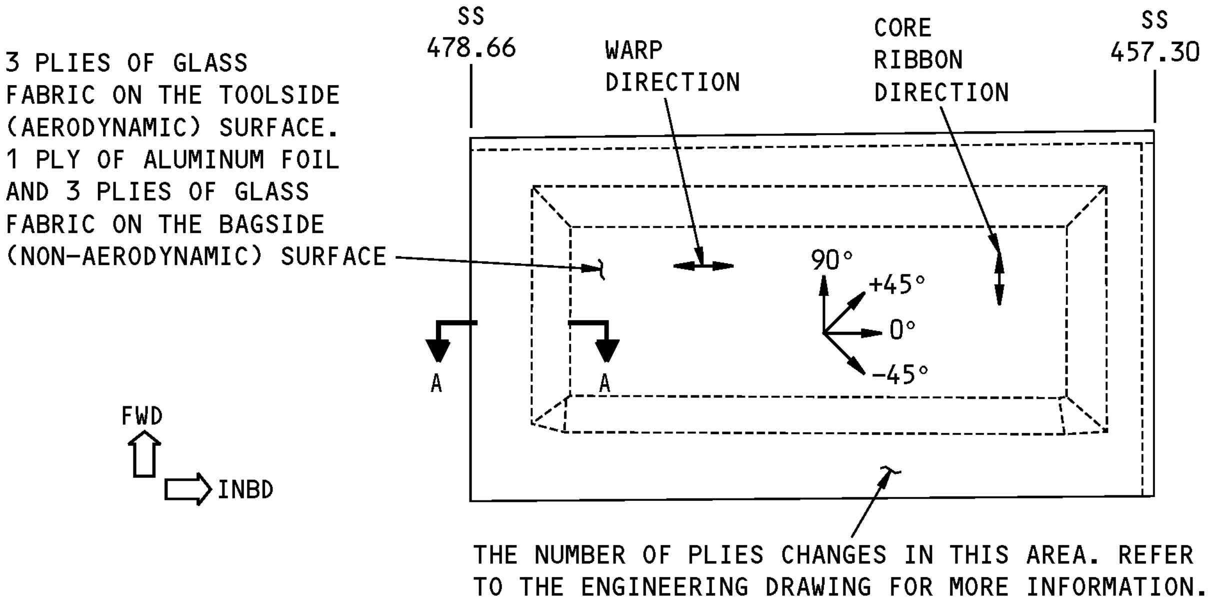
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 2, Item [2]**
**Figure 7**

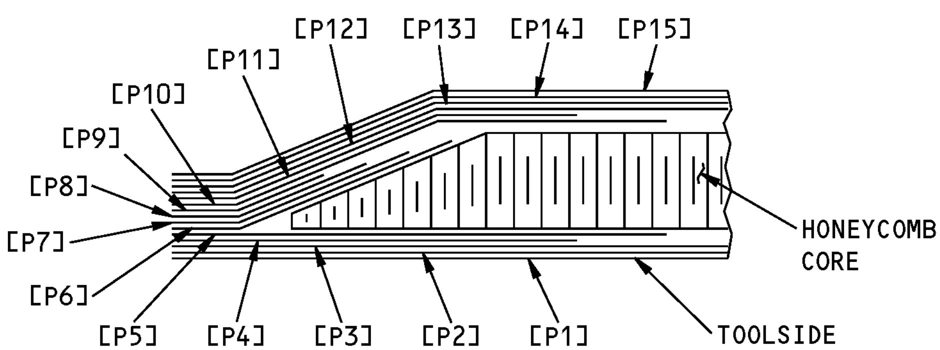
**Table 7**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 7 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Expanded aluminum Foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A (Optional: Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350) |
| P2 to P4, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P5, P6, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

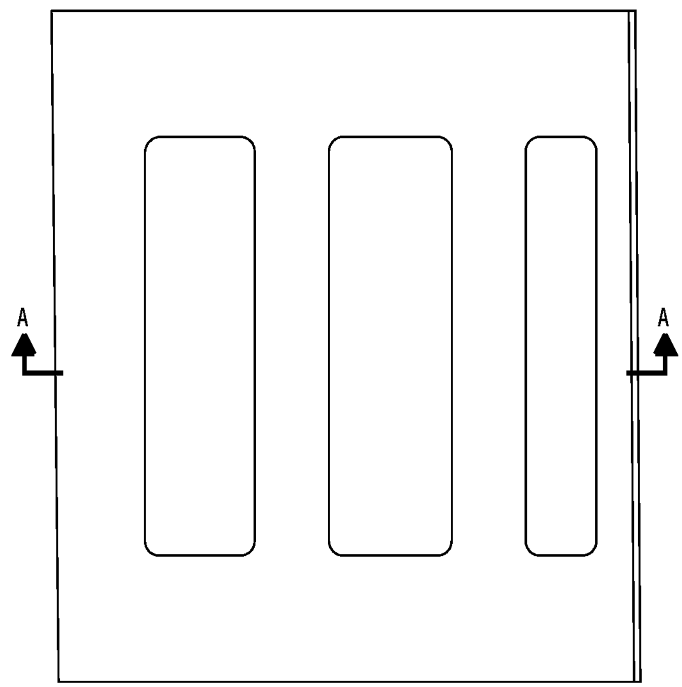
NOTES

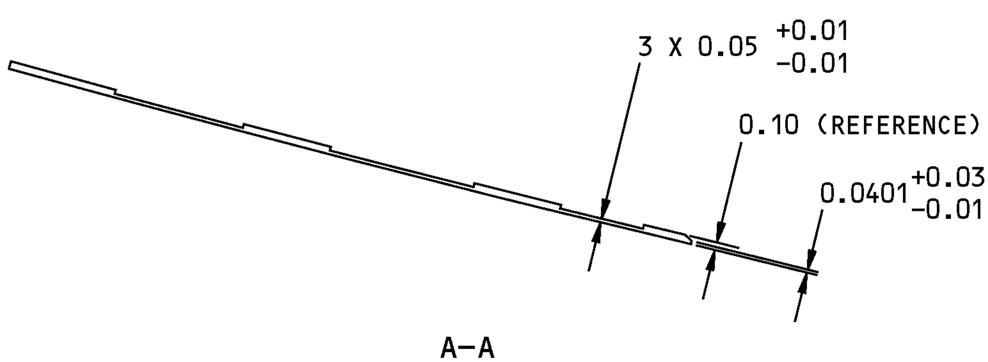
- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 2, Item [3]**
**Figure 8**

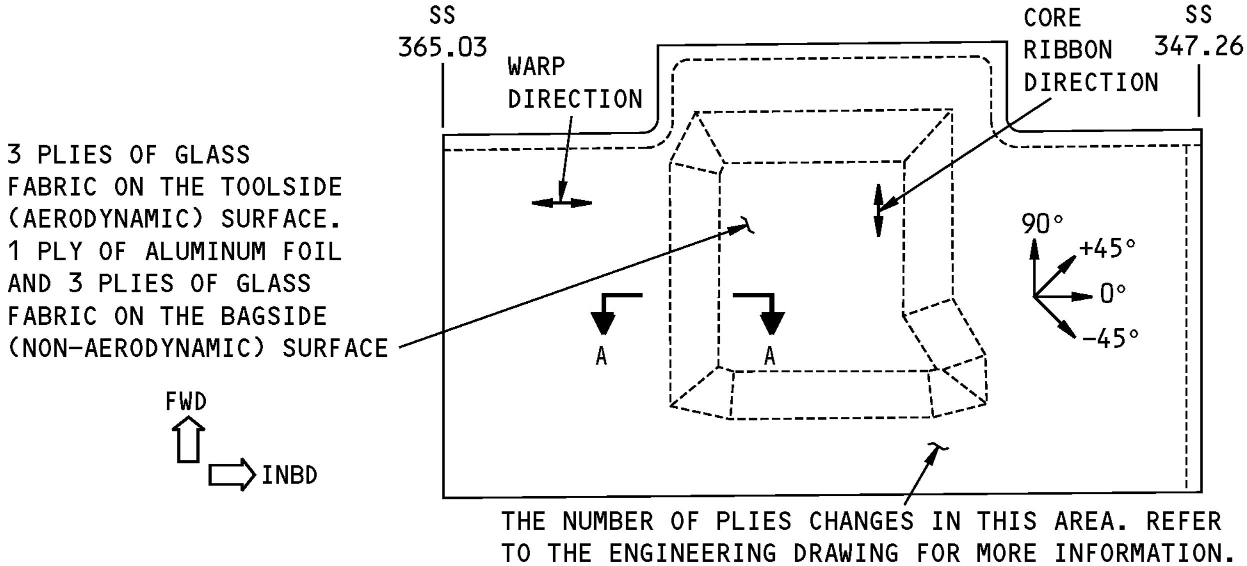
Table 8:

| PLY MATERIAL AND DIRECTION FOR FIGURE 8 | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |



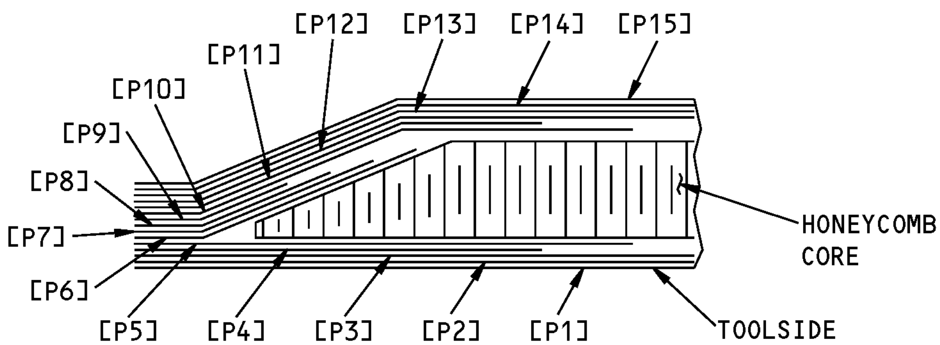




**Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 2, Item [4]**
**Figure 9**




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

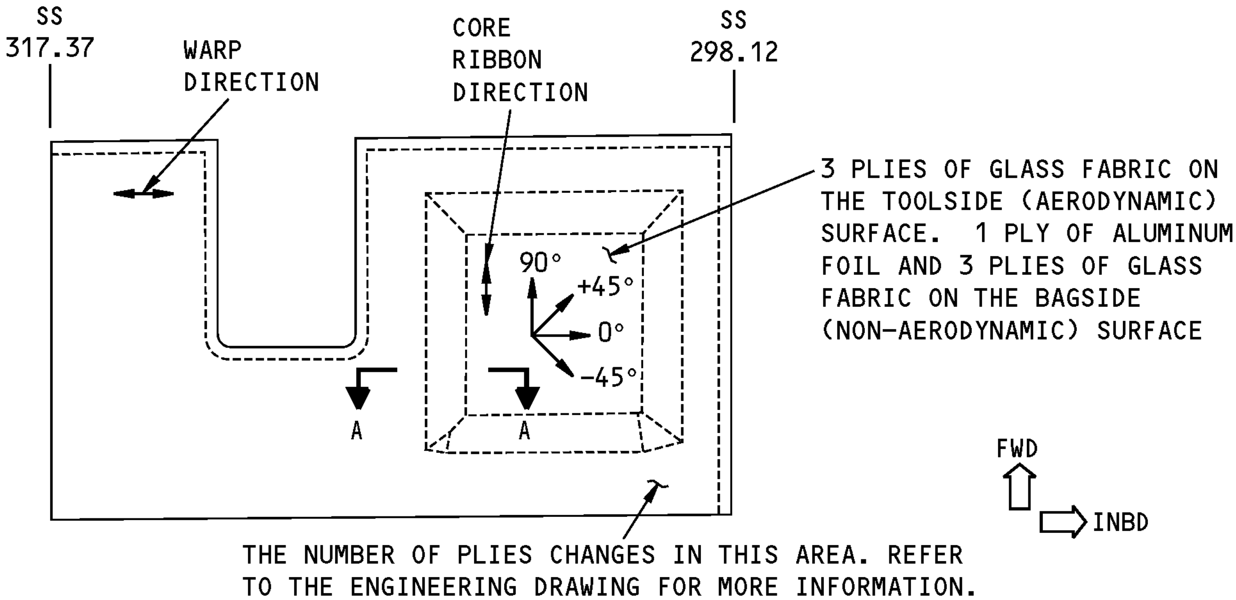
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 3, Item [1]**
**Figure 10**

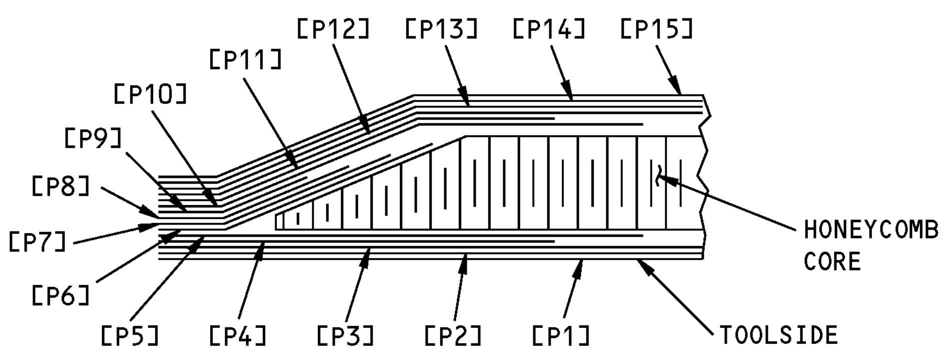
Table 9:

| PLY MATERIAL AND DIRECTION FOR FIGURE 10 | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

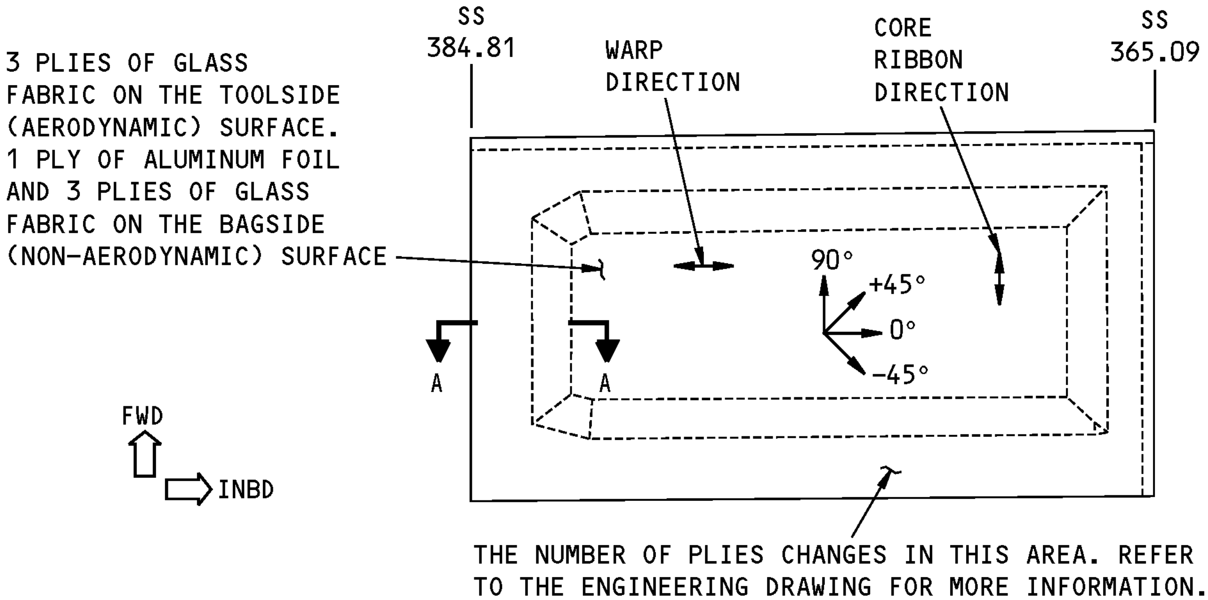
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 3, Item [2]**
**Figure 11**

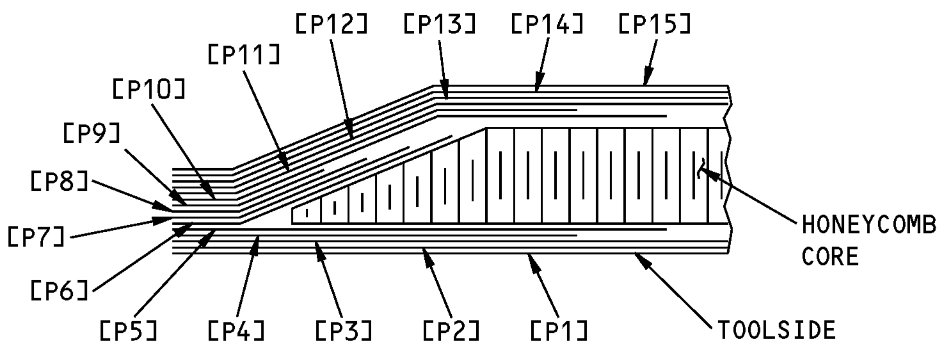
**Table 10**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 11 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

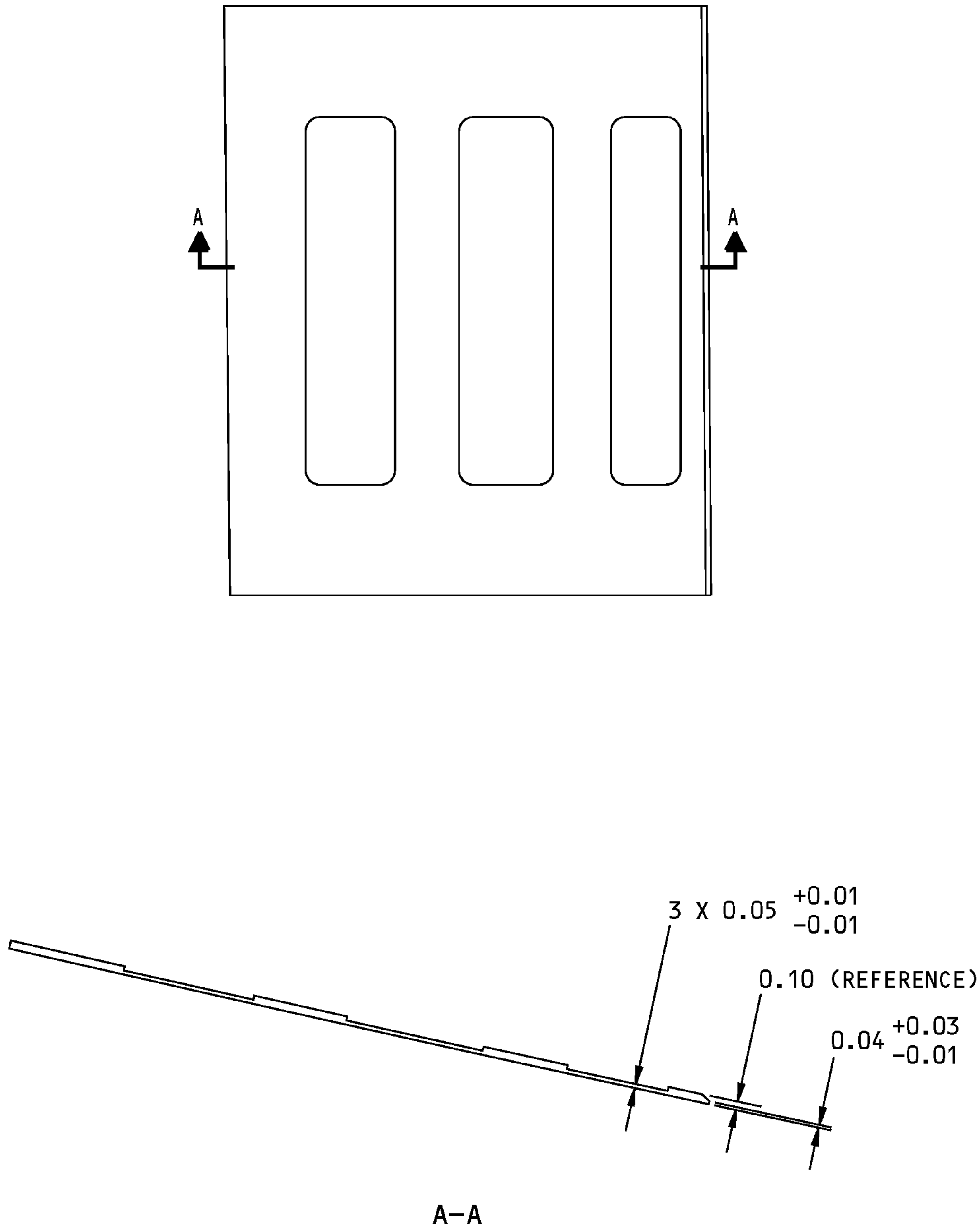
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 11 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

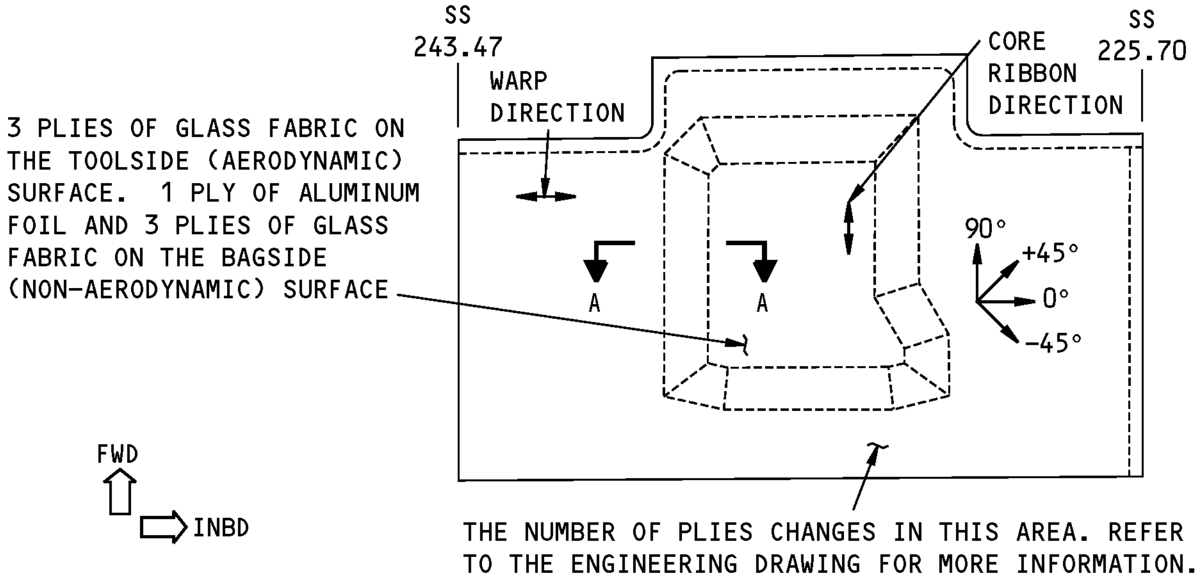
**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 3, Item [3]**
**Figure 12**

**Table 11**:

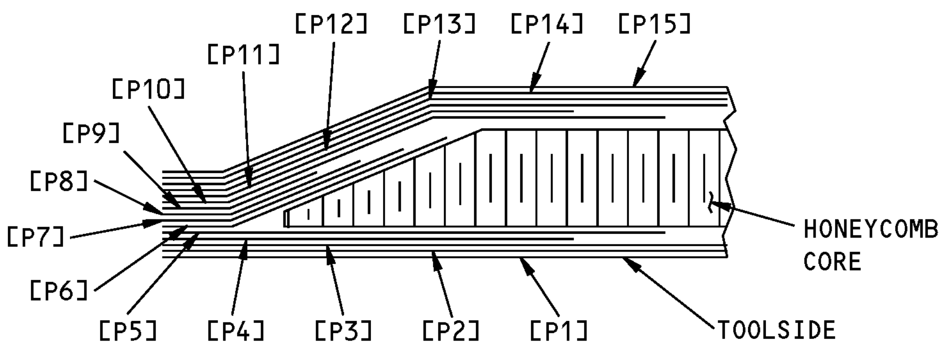
| PLY MATERIAL AND DIRECTION FOR FIGURE 12 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




**Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 3, Item [4]**
**Figure 13**




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION
A




A-A

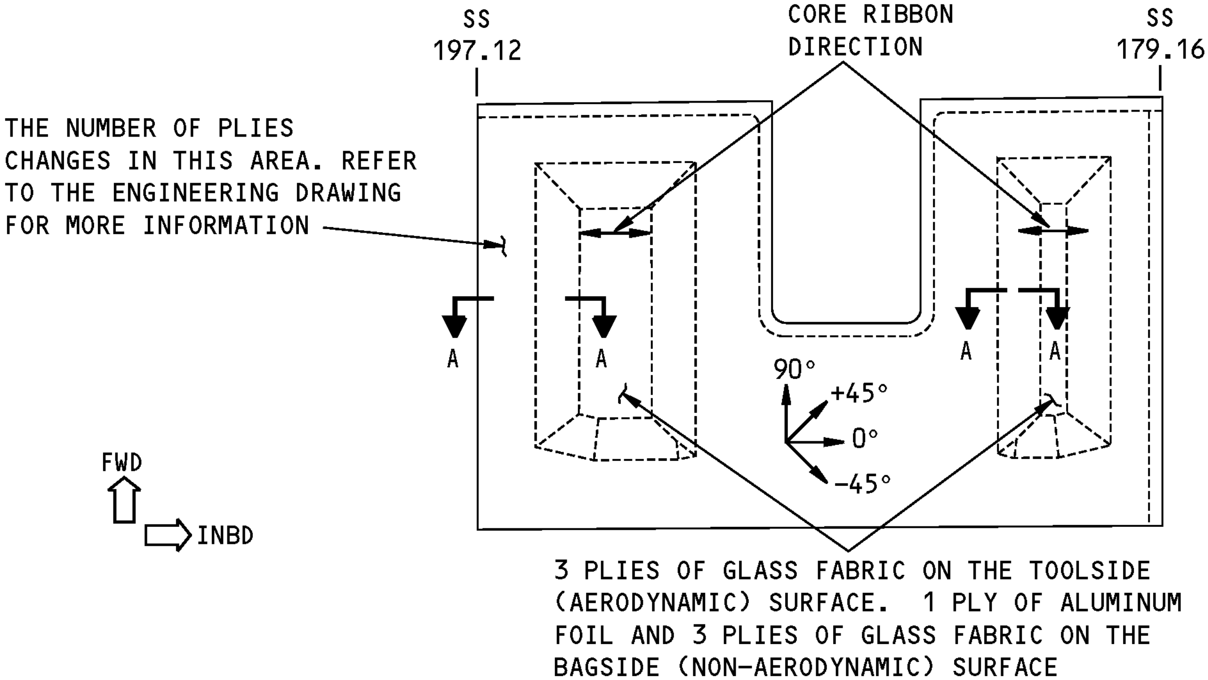
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 12 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 4, Item [1]**
**Figure 14**

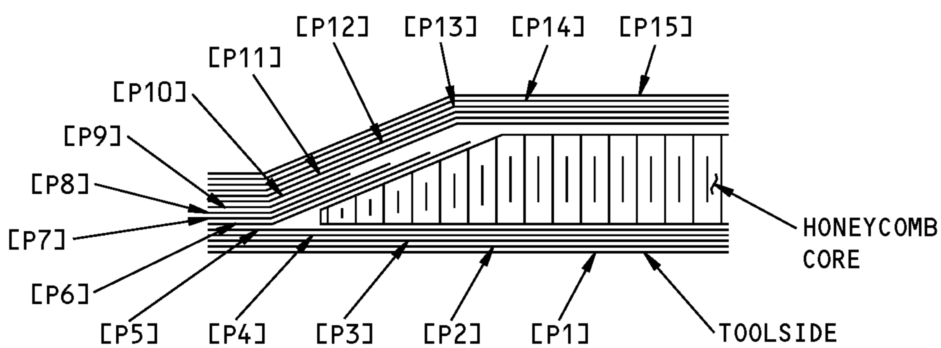
Table 12:

| PLY MATERIAL AND DIRECTION FOR FIGURE 14 | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

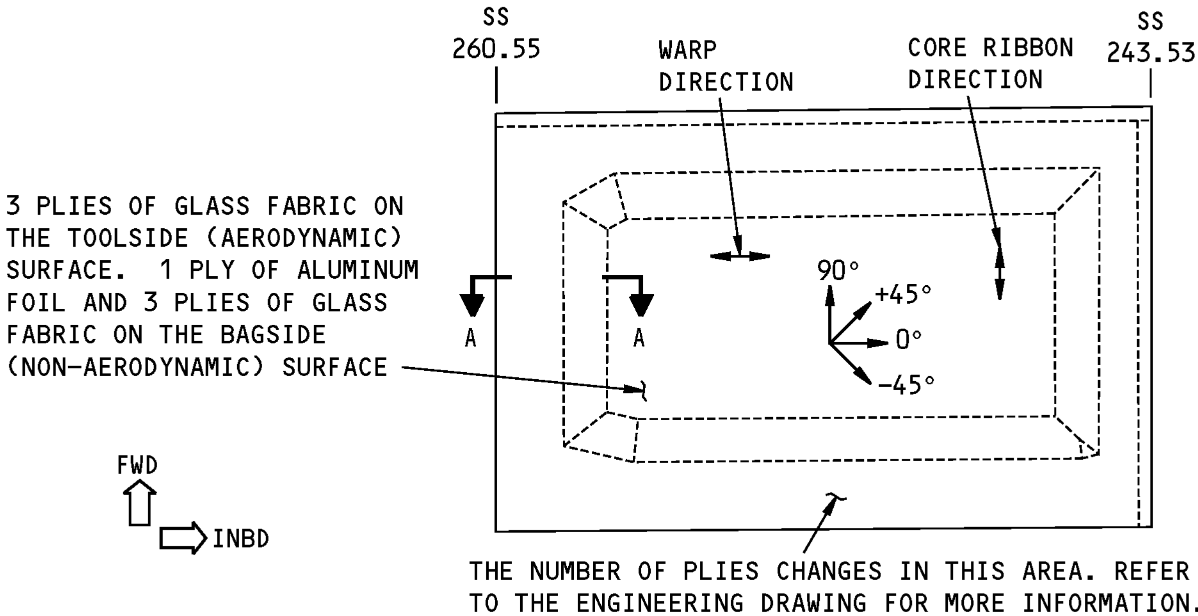
NOTES

- THE PLY DIRECTION IS THE SAME AS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 13 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 4, Item [2]**
**Figure 15**

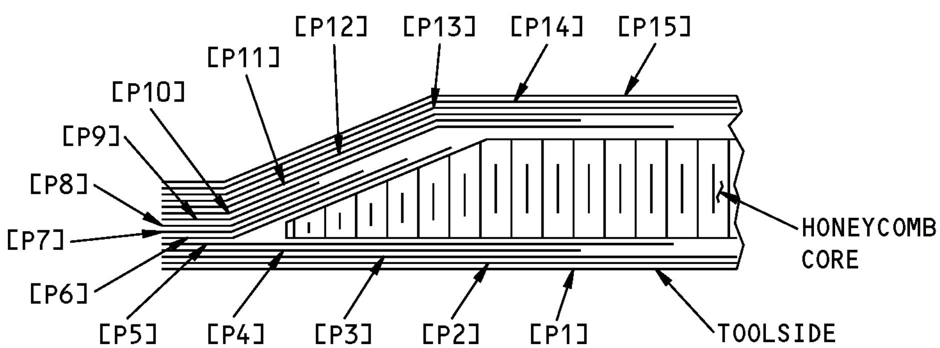
**Table 13**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 15 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

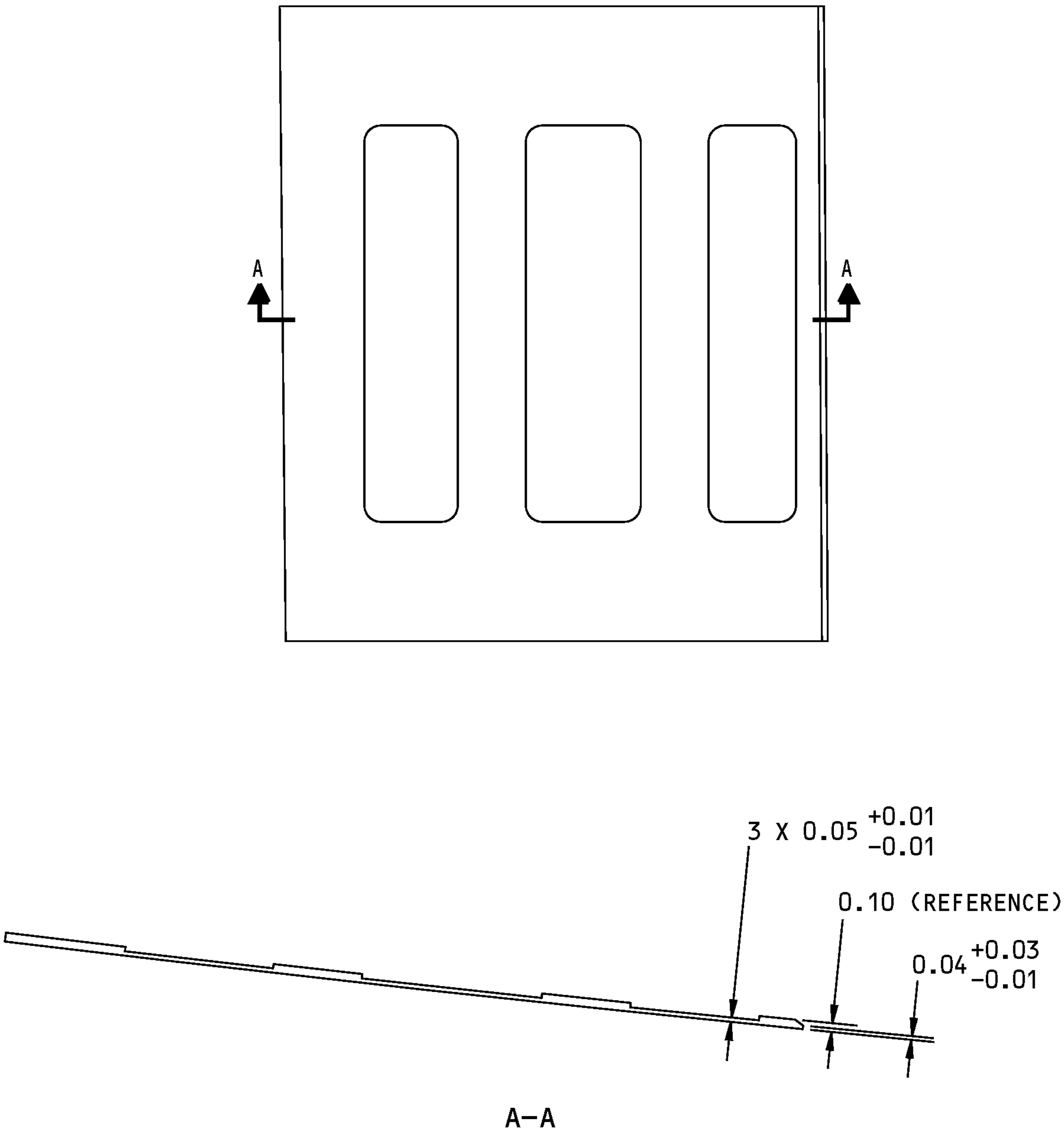
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 14 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

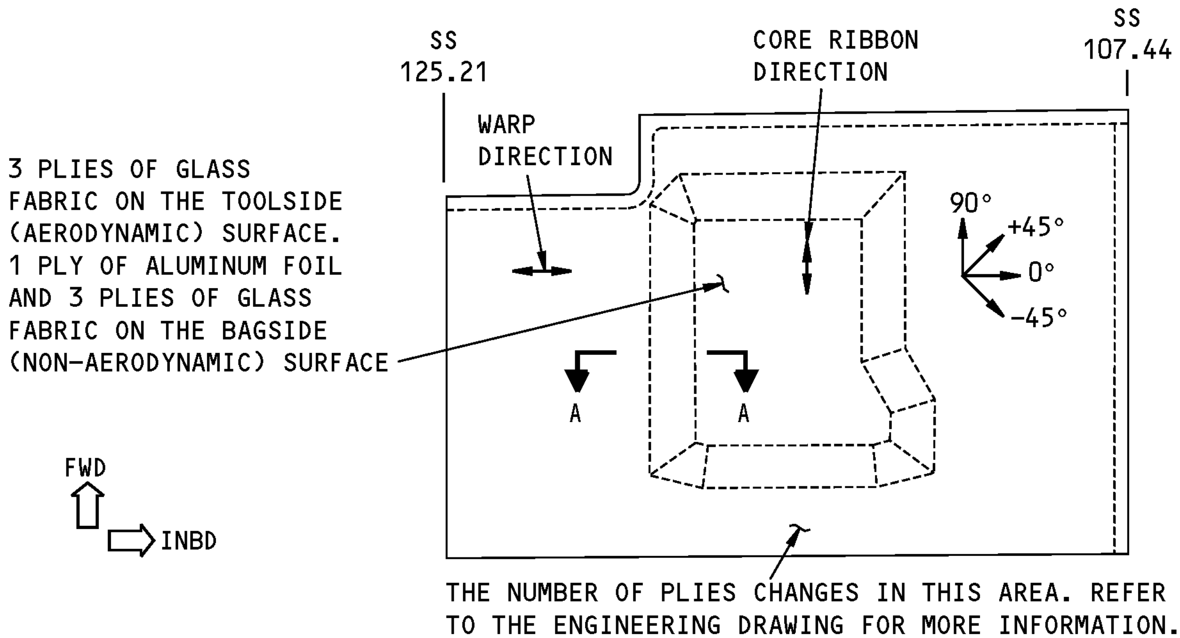
**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 4, Item [3]**
**Figure 16**

**Table 14**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 16 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ± 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ± 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |

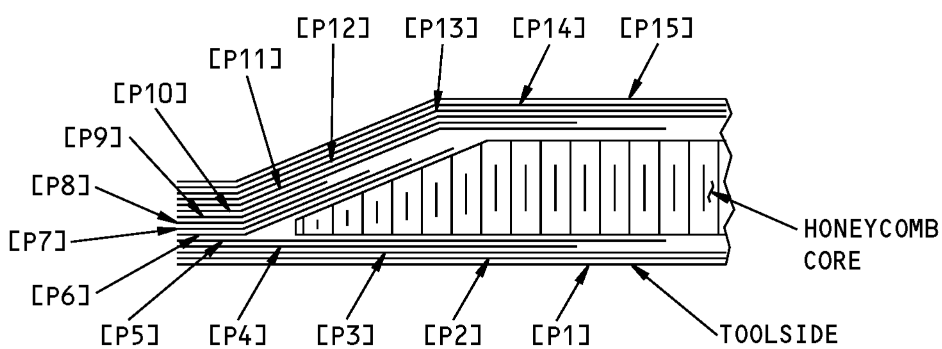



**Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 4, Item [4]**
**Figure 17**




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

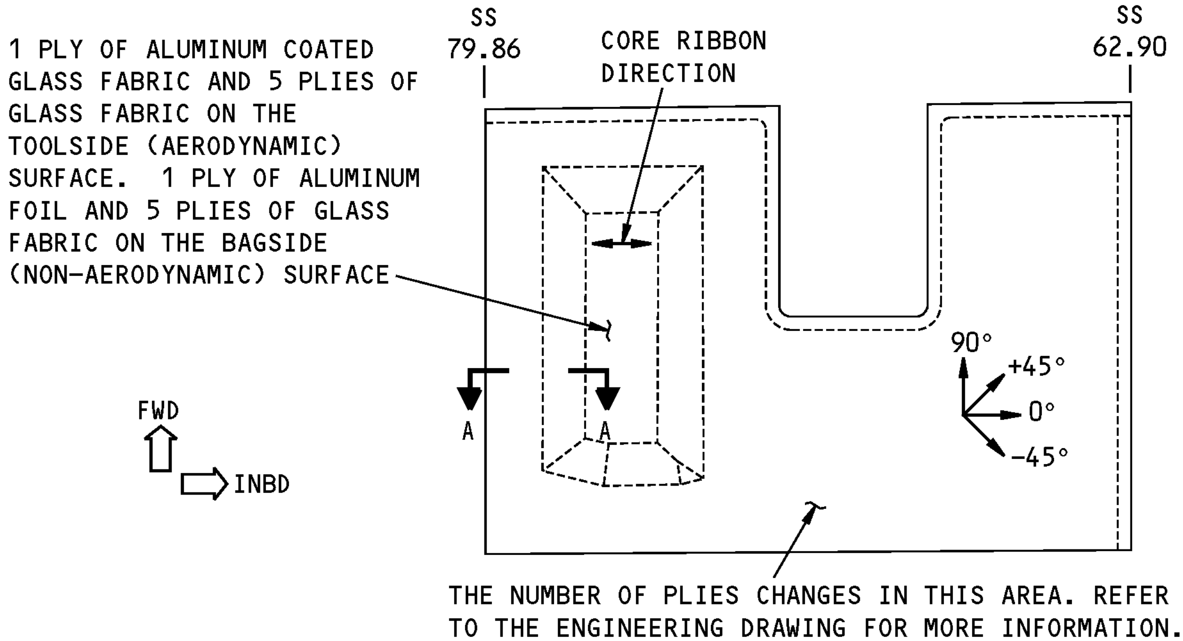
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 15 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [1]**
**Figure 18**

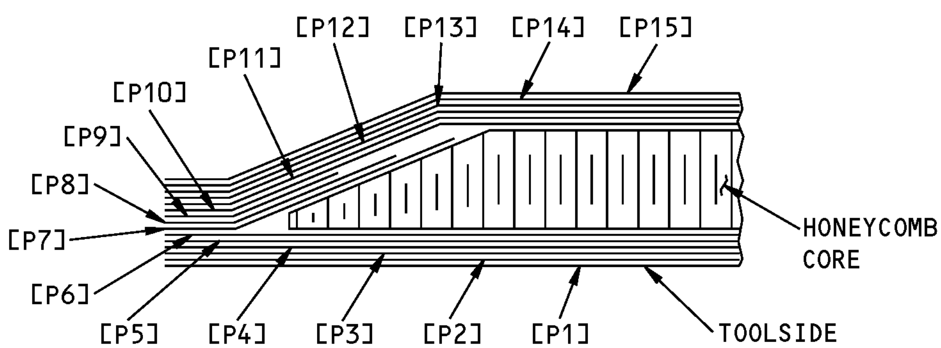
**Table 15**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 18 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

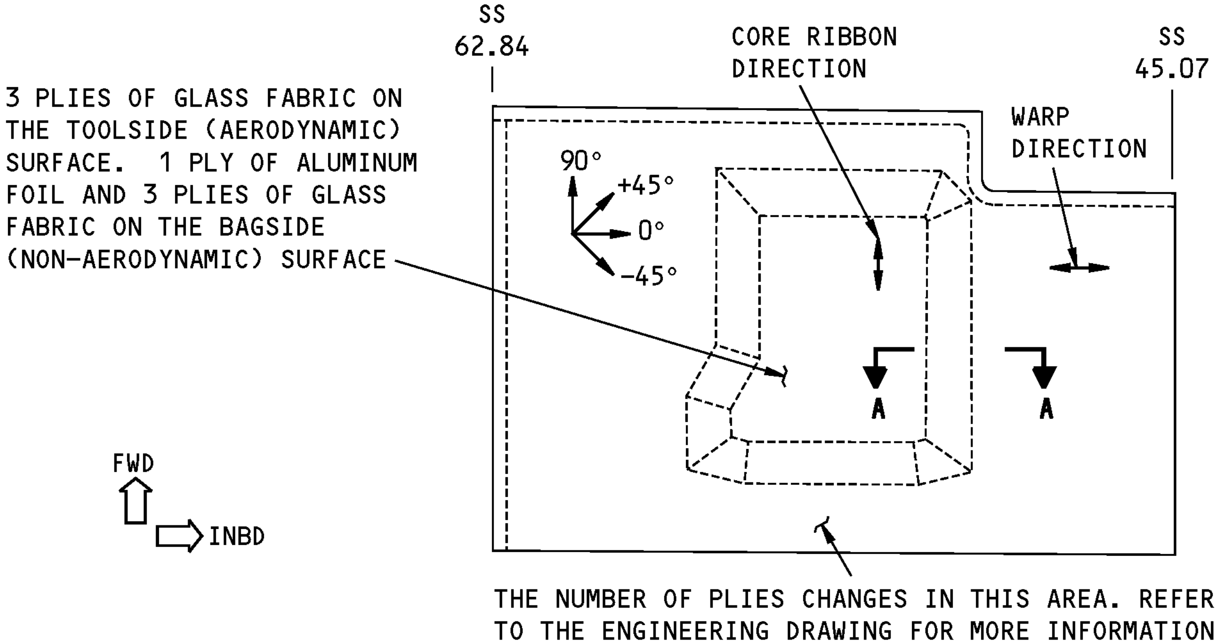
NOTES

- THE PLY DIRECTION IS THE SAME AS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 16 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [2]**
**Figure 19**

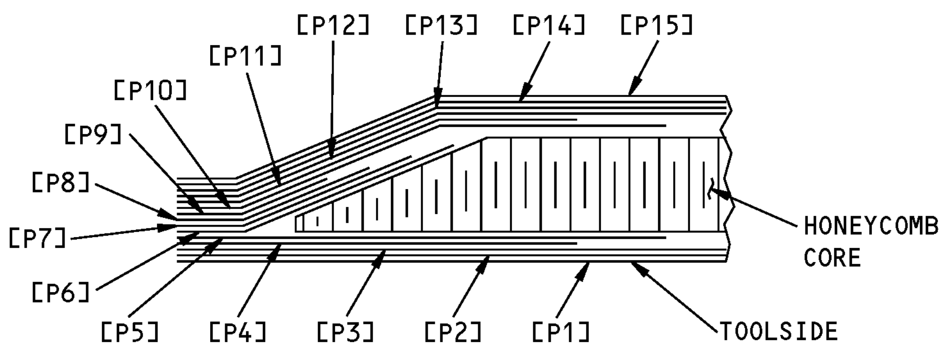
**Table 16**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 19 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I Class 1, Grade 016, Form A) |
| P2 to P4, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P5, P6, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

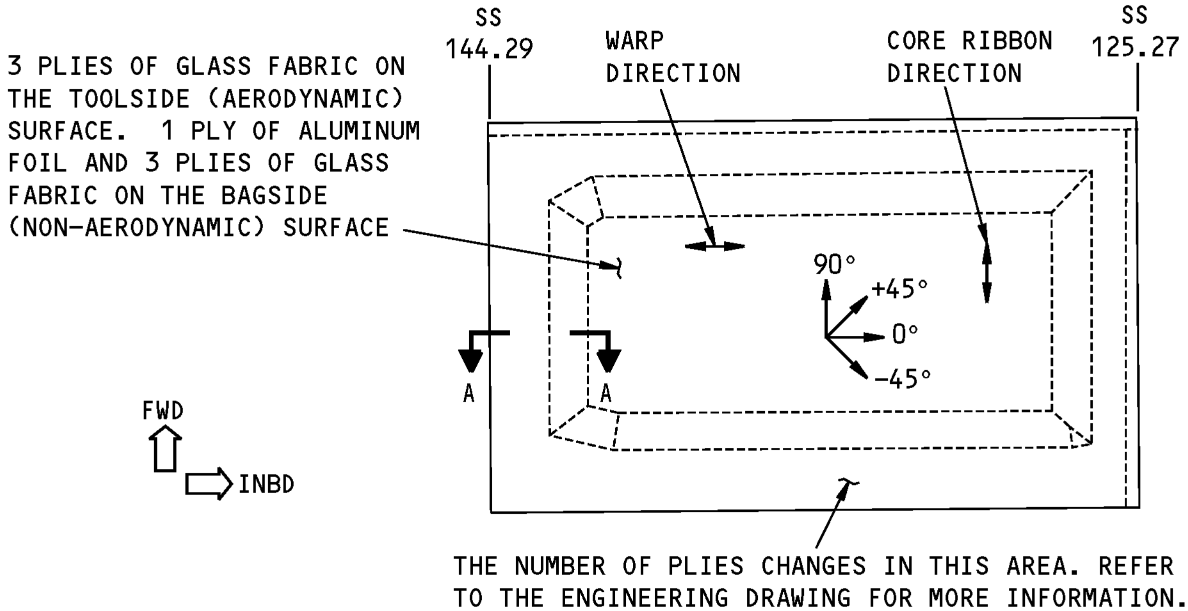
NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 17 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [3]**
**Figure 20**

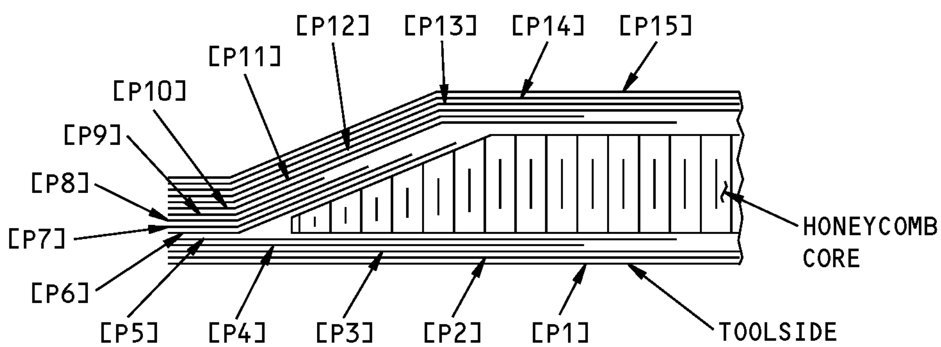
**Table 17**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 20 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I Class 1, Grade 016, Form A) |
| P2 to P4, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P5, P6, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

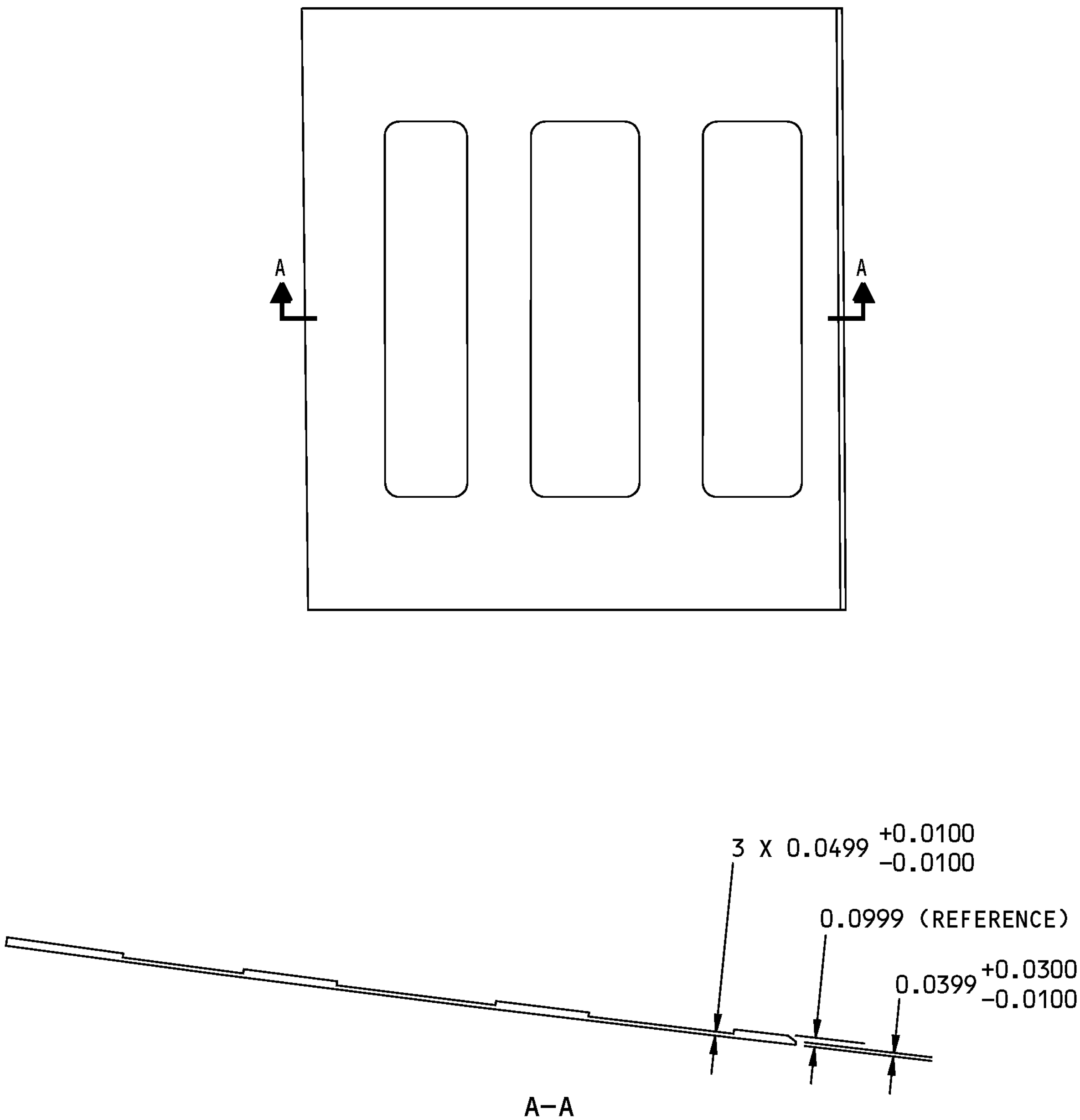
NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 18 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWINGS FOR SIMILAR PANELS IN THIS AREA.

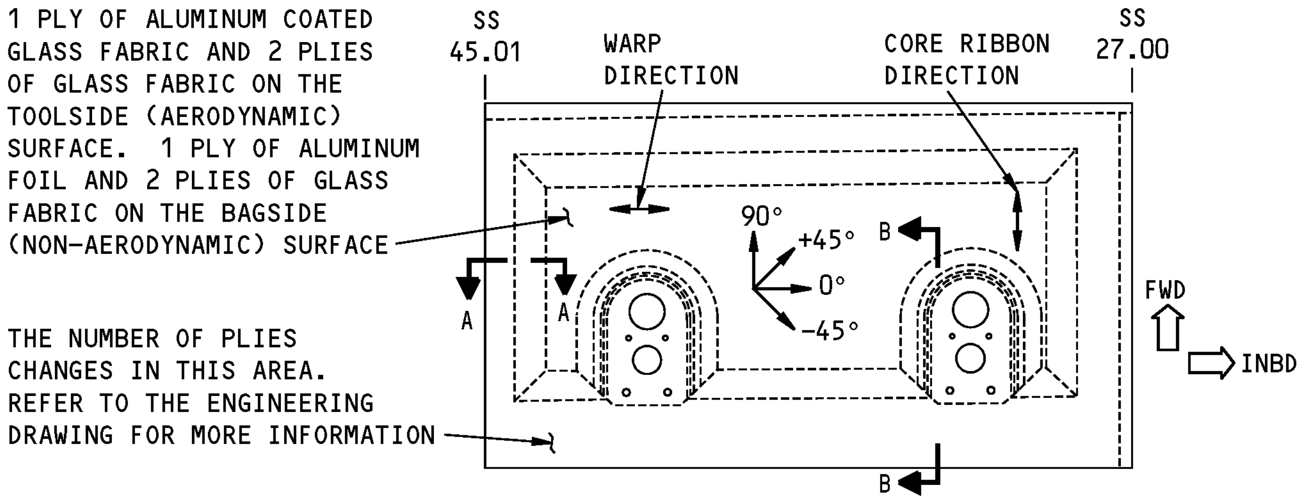
**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [4]**
**Figure 21**

Table 18:

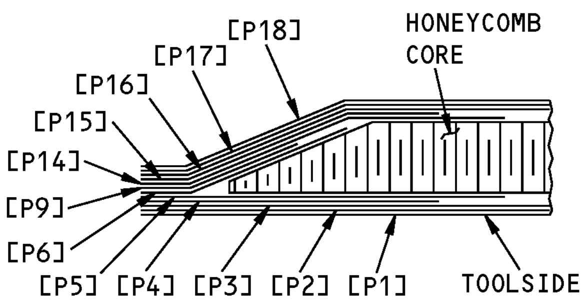
| PLY MATERIAL AND DIRECTION FOR FIGURE 21 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 to P3, and P12 to P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P4, P5, P10, and P11 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6, P7, and P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P8 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P15 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |



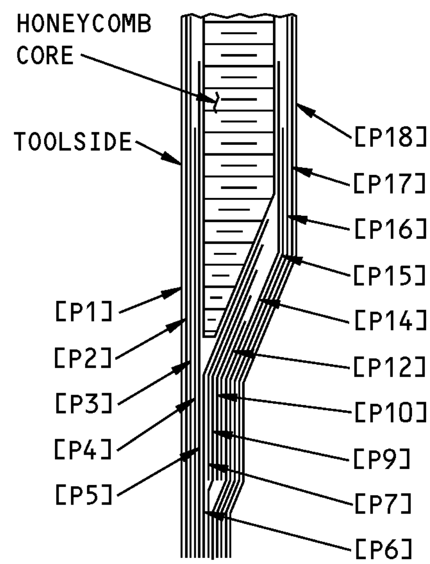

**Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 5, Item [5]**
**Figure 22**




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION
A




A-A




B-B

NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 19 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [6]**
**Figure 23**

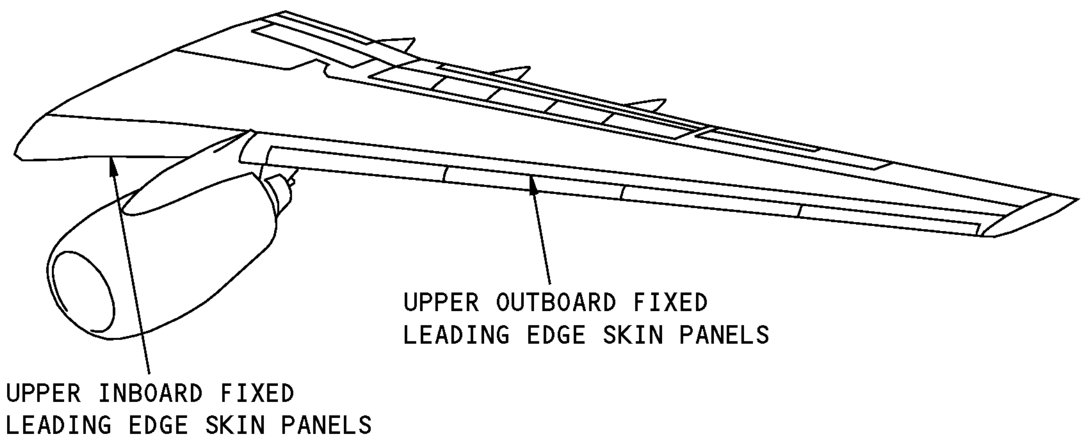
**Table 19**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 23 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | 0 or 90 degrees | Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I Class 1, Grade 016, Form A) |
| P2 to P5, P7, P8, and P10 to P17 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P6 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 |
| P9 | ±45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120 |
| P18 | ——— | Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B |

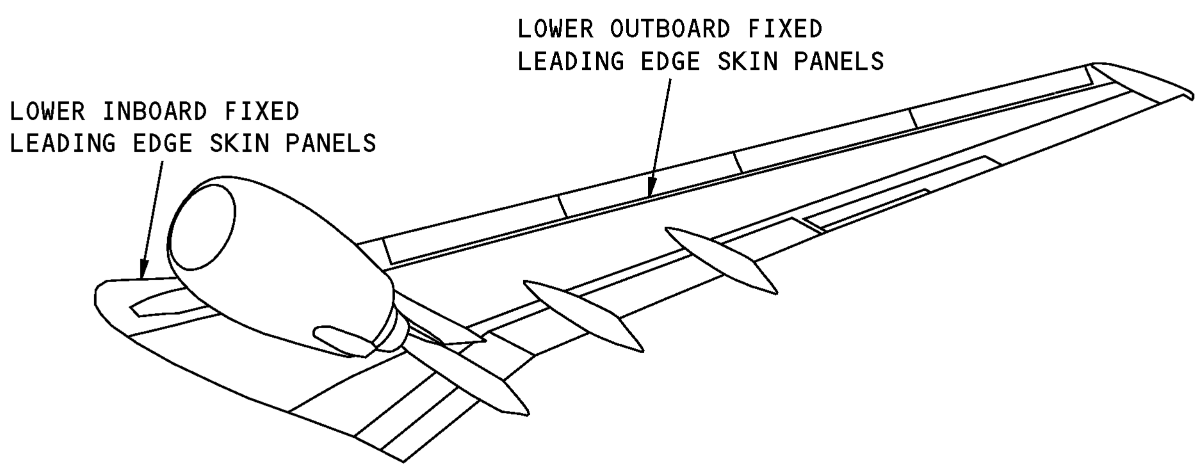
## ALLOWABLE DAMAGE 1 - WING FIXED LEADING EDGE PANELS

### 1. Applicability

A. Allowable Damage 1 is applicable to damage on the fixed leading edge panels shown in Wing Fixed Leading Edge Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Fixed Leading Edge Skin Panel Location**
**Figure 101**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limits that is applicable to each type of structure.

C. Do the steps that follow for the parts made of Glass Fiber Reinforced Plastic (GFRP).

(1) Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator, can be used.

(a) Refer to Definitions of the Damage Size, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(b) Refer to Definitions of the Facesheets, Figure 103/ALLOWABLE DAMAGE 1 for the definitions of the facesheets of a honeycomb core area.

(c) Refer to Wing Fixed Leading Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the GFRP parts.

(2) Remove all the contamination and water from the structure.

(a) Refer to 51-70-04 for the damage removal procedures.

(b) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| **TYPE OF STRUCTURE** | **PART NAME** | **PARAGRAPH** |
| ZONE 1 - GFRP/HONEYCOMB SANDWICH - EDGEBAND AND SOLID LAMINATE AREAS BETWEEN HONEY- COMB CORE AREAS - UPPER AND LOWER SKIN PANELS | UPPER AND LOWER PANELS | 4.A |
| | REMOVABLE PANEL | 4.A |
| | BEAM ASSEMBLY | 4.A |
| | ACCESS DOOR | 4.A |
| ZONE 2 - GFRP/HONEYCOMB SANDWICH - HONEYCOMB CORE AREAS OF THE UPPER SKIN PANELS | UPPER SKIN PANELS | 4.B |
| ZONE 3 - GFRP/HONEYCOMB SANDWICH - SOLID LAMINATE AREAS OF THE UPPER SKIN PANELS (DOES NOT INCLUDE SOLID LAMINATE AREAS IN ZONES 1) | UPPER PANELS | 4.C |
| ZONE 4 - GFRP/HONEYCOMB SANDWICH - HONEYCOMB CORE AREAS OF THE LOWER SKIN PANELS | LOWER PANELS | 4.D |
| | REMOVABLE PANEL | 4.D |
| | BEAM ASSEMBLY | 4.D |
| | ACCESS DOOR | 4.D |

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| TYPE OF STRUCTURE | PART NAME | PARAGRAPH |
| ALUMINUM | STRAKELETS | 4.E |
| | PANEL | 4.E |
| | OVERWING FAIRING | 4.E |
| | CLOSURE PANEL | 4.E |

(3) Some GFRP leading edge panels have a thin layer of aluminum on the outer surface.

(a) Refer to Wing Fixed Leading Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the GFRP parts with aluminum layers.

(b) The aluminum layer can be one of the three materials that follow:

1) BMS 8-278 aluminum coated glass fabric

2) BMS 8-289 bonded aluminum foil

3) BMS 8-336 expanded aluminum foil mesh.

(c) Refer to 51-70-14 for the allowable damage limits for the aluminum layer.

(4) Seal the damaged areas of the GFRP leading edge skin panels as follows:

(a) Seal all damaged areas that are not more than one ply deep (and that meet the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1 and 51-70-14) with one of the two procedures that follow:

1) Make a temporary seal.

a) Apply aluminum foil tape (speed tape).

b) Keep a record of the location.

c) Make sure the tape is in satisfactory condition at normal maintenance intervals.

2) Make a permanent seal.

a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

b) Apply one layer of BMS 10-103, Type I or BMS 10-79, Type III primer. Refer to SOPM 20-44-04 for the application of BMS 10-79, Type III primer.

c) Apply one layer of BMS 10-60 enamel. Refer to AMM PAGEBLOCK 51-21-99/701.

(b) Seal all damaged areas that are more than one ply deep (and that meet the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1 and 51-70-14) as follows:

1) Use a vacuum and heat to remove moisture from the solid laminate and/or honeycomb cells. Refer to 51-70-04.

2) Make a temporary seal with aluminum foil tape (speed tape).

3) Keep a record of the location.

4) Repair the damage permanently no later than 24 months from the time the seal was made.

D. Do the steps that follow for the aluminum parts.

(1) Refer to Wing Fixed Leading Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the aluminum parts.

(2) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of the abrasive and other materials you need to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

(d) Make the surface texture roughness for all cut surfaces 125 microinches Ra or smoother.

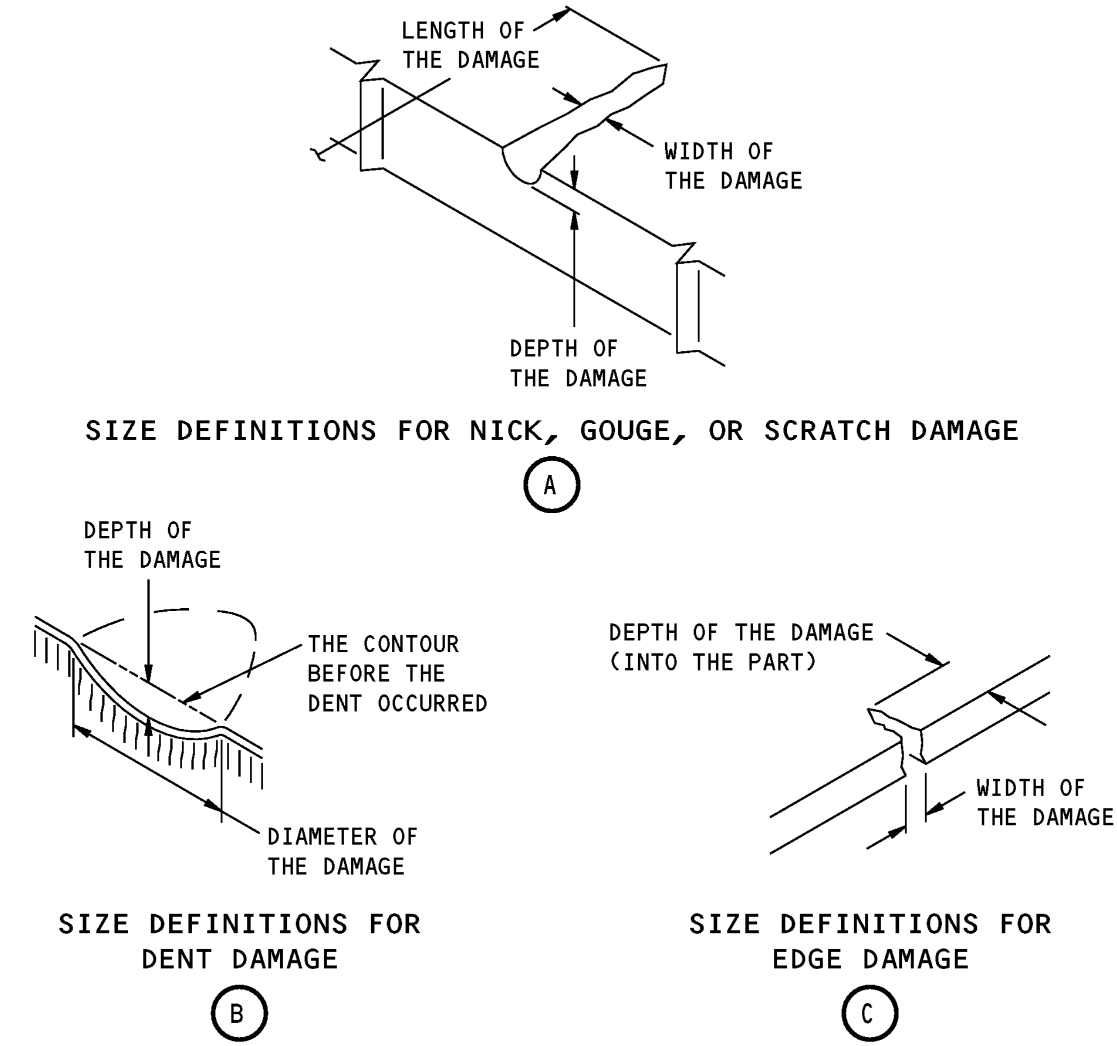
(3) After you remove the damage on the internal surfaces, do the steps that follow:

(a) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

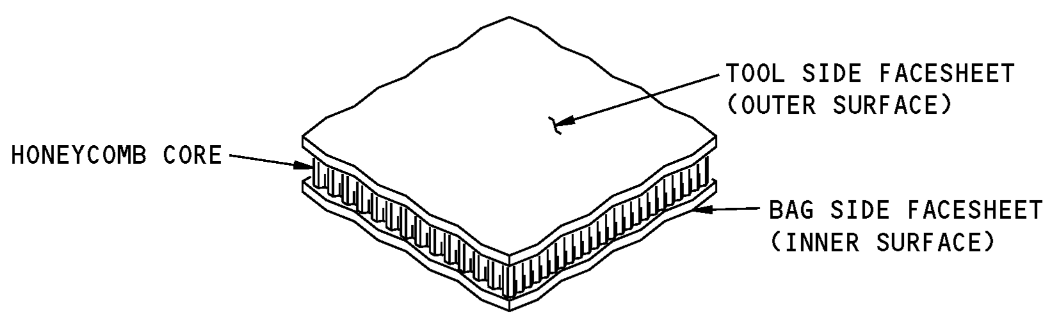
(b) Apply one layer of BMS 10-11, Type I primer to the conversion coated, reworked areas. Refer to SOPM 20-41-02.

(4) After you remove the damage on the external surfaces, do the step that follows:

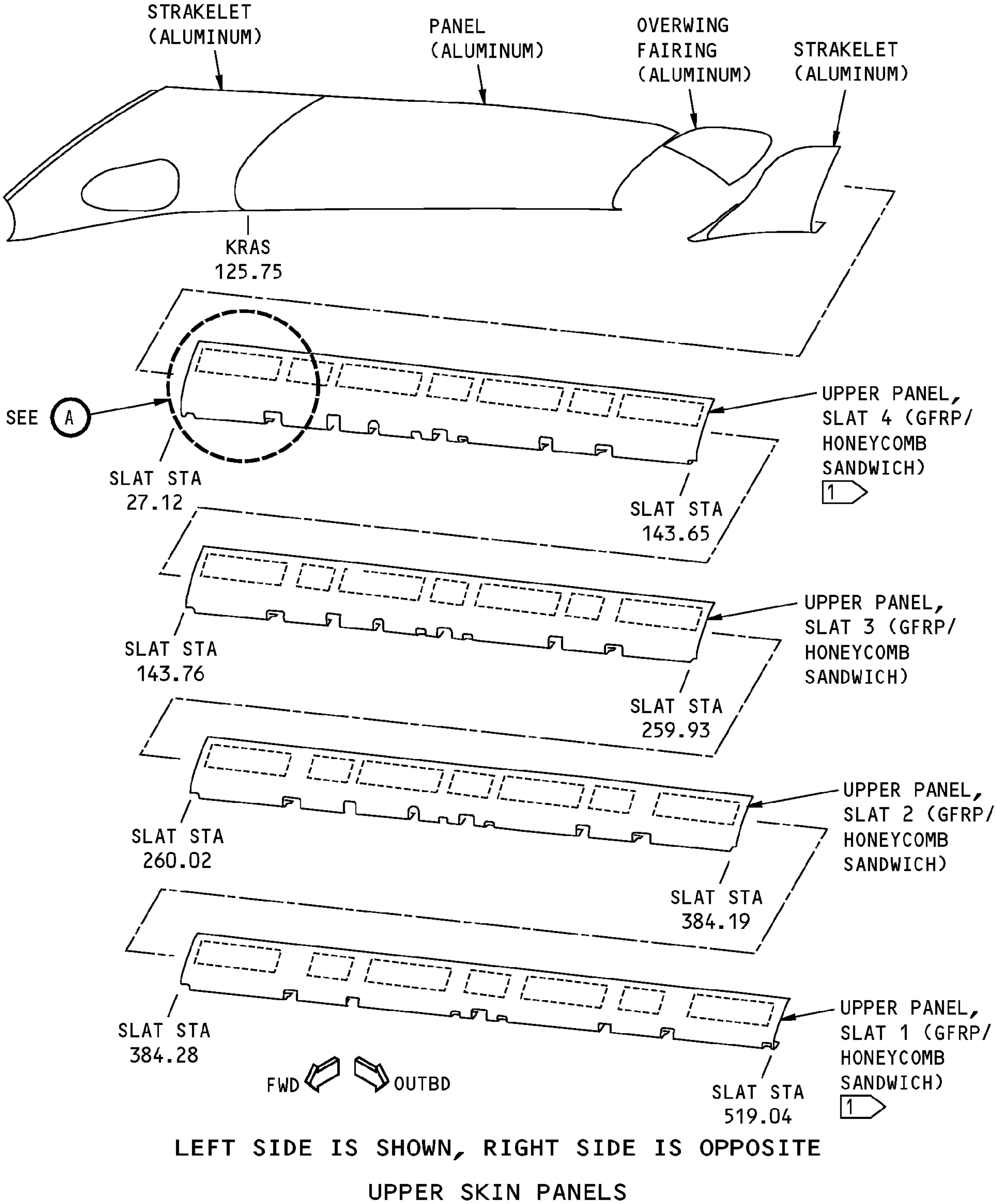
(a) Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.



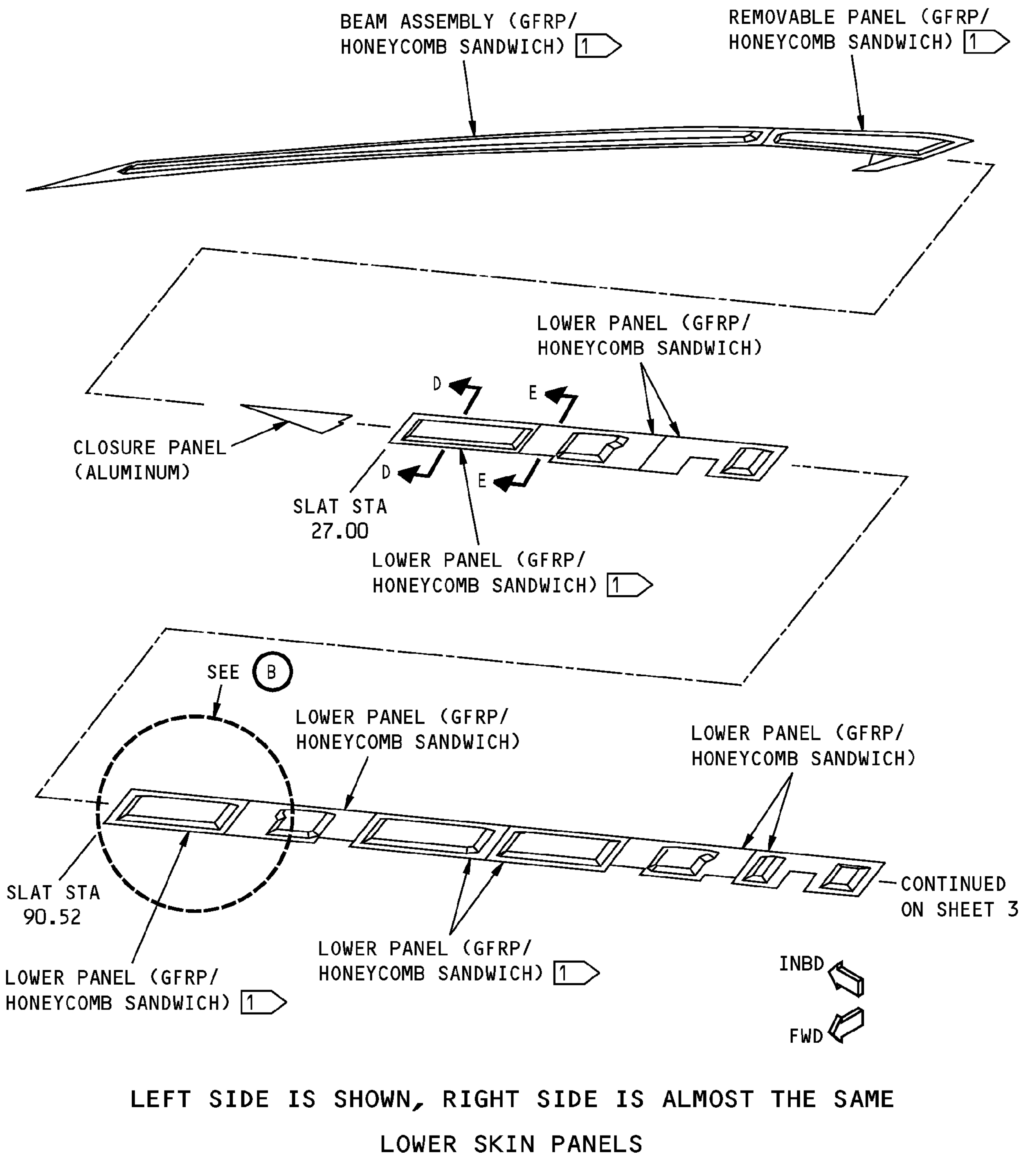

**Definitions of the Damage Size**
**Figure 102**



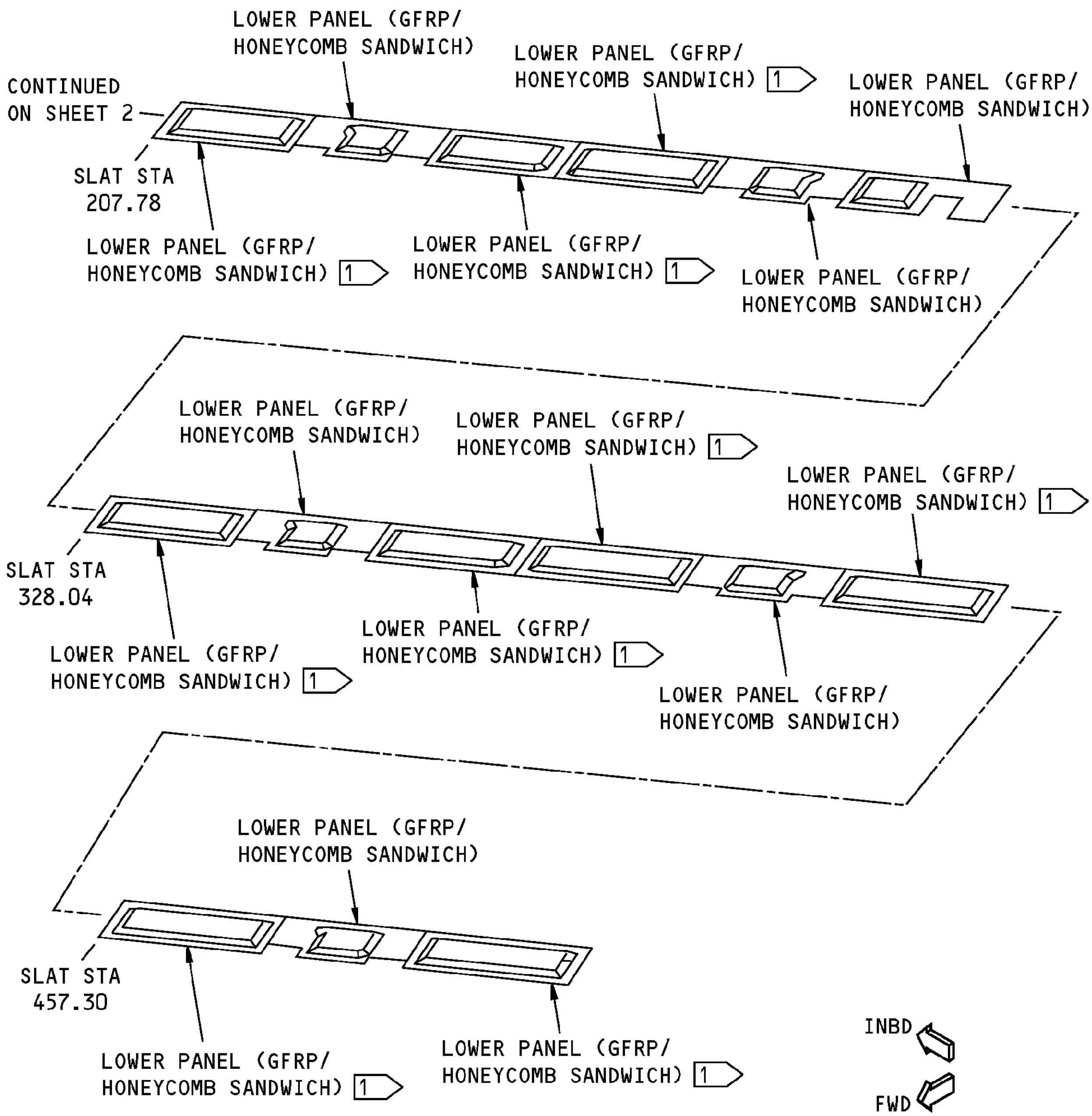

**Definitions of the Facesheets**
**Figure 103**




**Wing Fixed Leading Edge Skin Allowable Damage**
**Figure 104 (Sheet 1 of 5)**



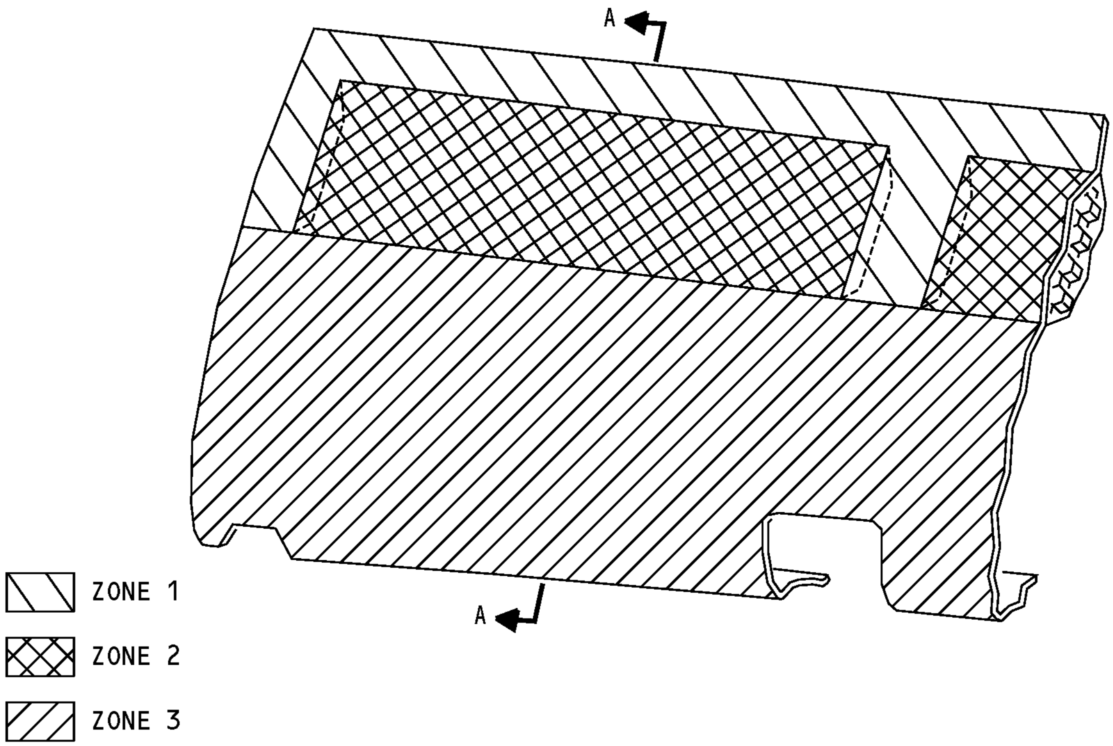

**Wing Fixed Leading Edge Skin Allowable Damage**
**Figure 104 (Sheet 2 of 5)**



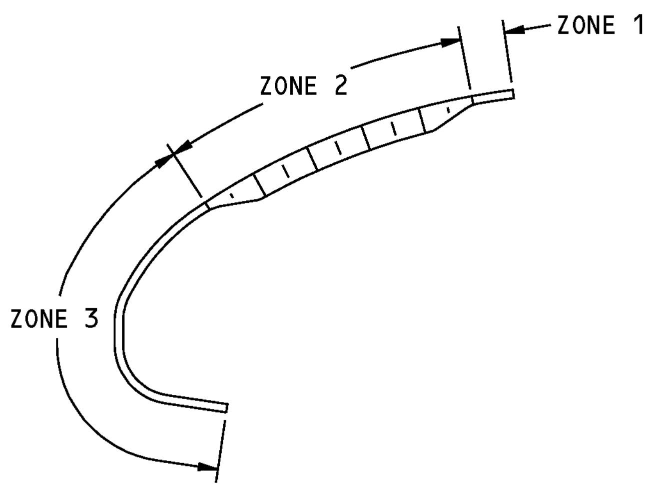

LEFT SIDE IS SHOWN, RIGHT SIDE IS ALMOST THE SAME

LOWER SKIN PANELS

**Wing Fixed Leading Edge Skin Allowable Damage**
**Figure 104 (Sheet 3 of 5)**

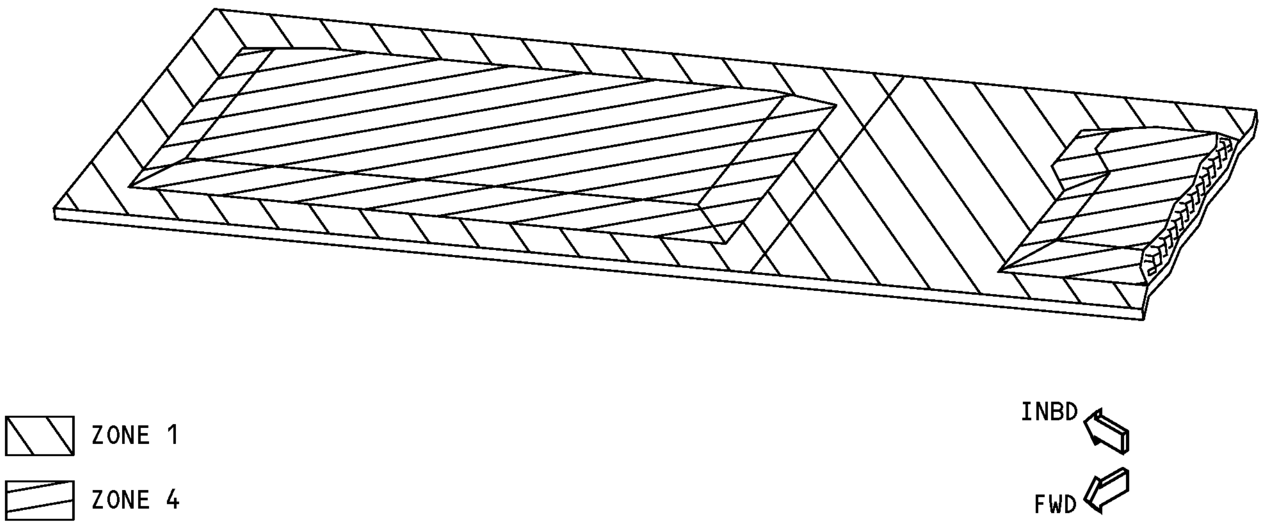



ALLOWABLE DAMAGE ZONES
(TYPICAL FOR ALL UPPER PANELS)
A




A-A
(TYPICAL)

**Wing Fixed Leading Edge Skin Allowable Damage**
**Figure 104 (Sheet 4 of 5)**




ALLOWABLE DAMAGE ZONES
(TYPICAL FOR GFRP LOWER SKIN PANELS)
B

NOTES

- ALL GFRP/HONEYCOMB SANDWICH PANELS HAVE BMS 8-289 BONDED ALUMINUM FOIL ON THE LOWER SURFACE.
- ZONE 1: EDGEBAND AND SOLID LAMINATE AREAS BETWEEN HONEYCOMB CORE AREAS
- ZONE 2: HONEYCOMB AREAS OF THE UPPER SKIN PANELS
- ZONE 3: SOLID LAMINATE AREAS (DOES NOT INCLUDE THE SOLID LAMINATE AREA OF ZONE 1)
- ZONE 4: HONEYCOMB CORE AREAS OF THE LOWER SKIN PANELS

1 BMS 8-336 EXPANDED ALUMINUM FOIL MESH OR BMS 8-278 ALUMINUM COATED GLASS FABRIC ON THE UPPER SURFACE.

**Wing Fixed Leading Edge Skin Allowable Damage**
**Figure 104 (Sheet 5 of 5)**

### 3. References

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| 51-70-14 | STRUCTURES WITH ALUMINUM COATINGS AND FOILS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

### 4. Allowable Damage Limits

A. Zone 1: GFRP/honeycomb Sandwich - Edgebands and Solid Laminate Area Between Honeycomb Core Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 1.0 inch in length

(c) A maximum of 0.25 inch in width

(d) A minimum of 0.50 inch away from the edge of a fastener hole

(e) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

**Table 102**:

| MINIMUM SPACING DIMENSIONS | | |
|---|---|---|
| ZONE | PART NAME | MINIMUM DAMAGE SPACING "A" (INCHES) |
| Zone 1 | Upper Panels, Lower Panels, Beam Assembly, Access Door, Removable Panels | 1.5x(D + d) |
| Zone 2 | Upper Panel, Slat 1 | 1.5x(D + d) |

| MINIMUM SPACING DIMENSIONS | | |
|---|---|---|
| ZONE | PART NAME | MINIMUM DAMAGE SPACING "A" (INCHES) |
| | Upper Panel, Slat 2, Upper Panel, Slat 3, Upper Panel, Slat 4 | 1.5x(D+d) |
| Zone 3 | Upper Panels | 1.5x(D+d) |
| Zone 4 | Lower Panels, Beam Assembly, Access Door, Removable Panels | 1.5x(D+d) |

(3) Dents that do not cause damage to the glass fibers are permitted if they are:

(a) A maximum of 0.625 inch in diameter

(b) A maximum of two plies in depth

**NOTE**: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 2 plies.

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.625 inch in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch from the edge of a fastener hole.

(5) Delaminations are permitted if they are:

(a) A maximum of 0.625 inch in diameter

(b) A minimum of 0.50 inch away from the edge of a fastener hole

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(6) Edge Erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ALLOWABLE DAMAGE 1.

B. Zone 2: GFRP/Honeycomb Sandwich - Honeycomb Core Areas of the Upper Skin Panels

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 5.0 inches in length

(c) A minimum of 0.5 inch from the edge of a fastener hole

(d) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A maximum of 2 plies in depth

**NOTE**: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 2 plies.

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 0.5 inch away from a fastener hole

(b) A maximum of 2.0 inches in diameter

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch away from a fastener hole.

C. Zone 3: GFRP/Honeycomb Sandwich - Solid Laminate Areas (does not include the edgebands and solid laminate areas in Zone 1)

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 1.0 inch in length

(c) A maximum of 0.25 inch in width

(d) A minimum of 0.50 inch away from the edge of a fastener hole

(e) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 1.0 inch in diameter

(b) A maximum of 2 plies in depth

**NOTE**: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 2 plies.

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 1.0 inch in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch away from a fastener hole.

(5) Delaminations are permitted if they are:

(a) A maximum of 1.0 inch in diameter

(b) A maximum depth of 0.10 inch from an edge

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(6) Edge Erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ALLOWABLE DAMAGE 1.

D. Zone 4: GFRP/Honeycomb Sandwich - Honeycomb Core Areas of the Lower Skin Panels

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 5.0 inches in length

(c) A minimum of 0.5 inch from the edge of a fastener hole

(d) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 2 plies in depth

**NOTE**: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 0.10 inch.

(b) A maximum of 2.0 inches in diameter

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 0.5 inch away from a fastener hole

(b) A maximum of 2.0 inches in diameter

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies
2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies
2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch away from a fastener hole.

E. Aluminum - Strakelet, Panel, Overwing Fairing, and Closure Panel

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

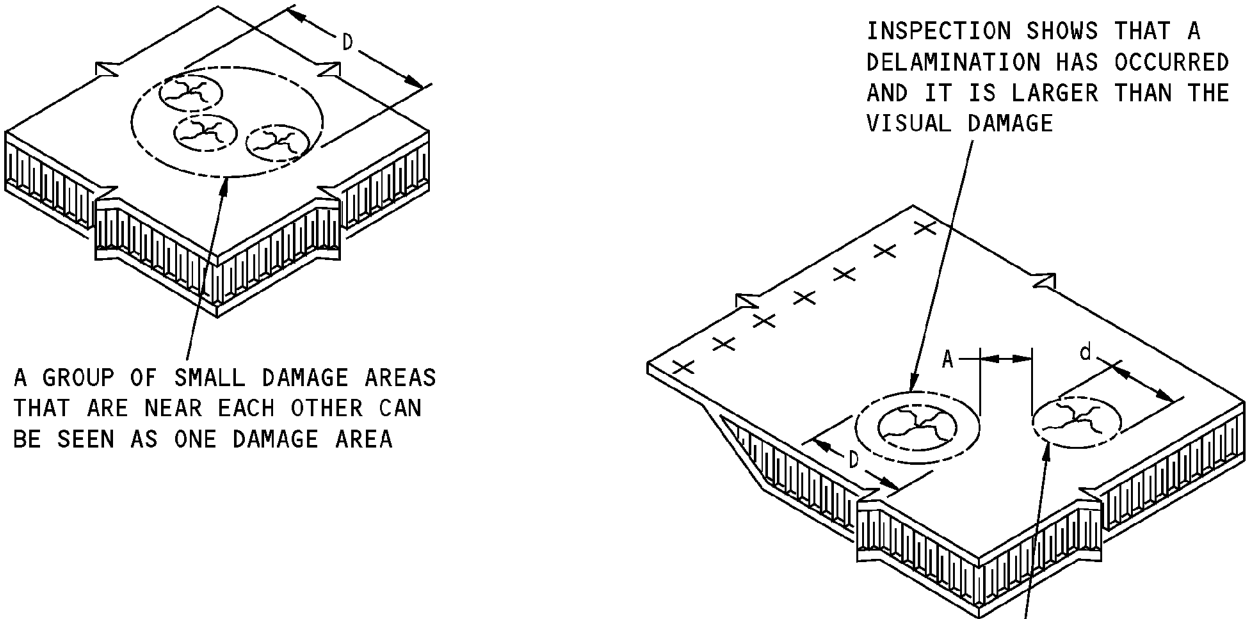
(3) Dents are permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail F .

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 1.0 inch away from the edge of a hole, other damage, or material edge

(b) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

1) Use a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.
2) Drill the holes to the applicable size as given in 51-40-05.



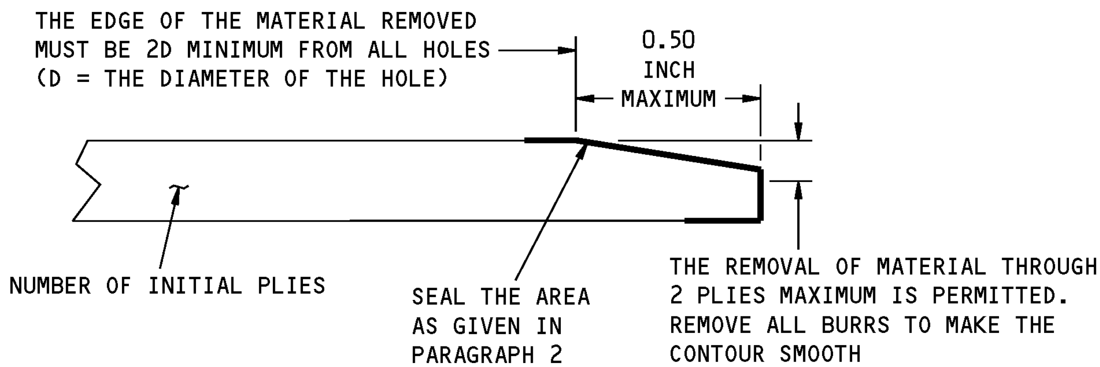

NOTE: THE DIMENSION OF A DAMAGE AREA IS EITHER THE DIMENSION OF THE VISUAL DAMAGE OR THE DIMENSION OF THE DELAMINATION. USE THE DIMENSION OF THE LARGER DAMAGE.

D IS THE LARGER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

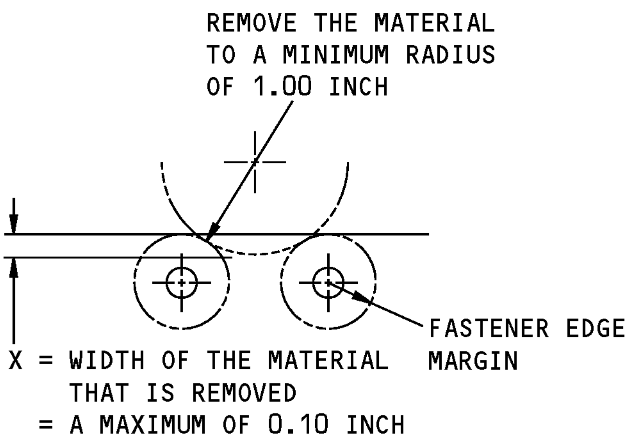
d IS THE SMALLER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

A IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

**Damage that is Permitted to Honeycomb Core Areas**
**Figure 105**

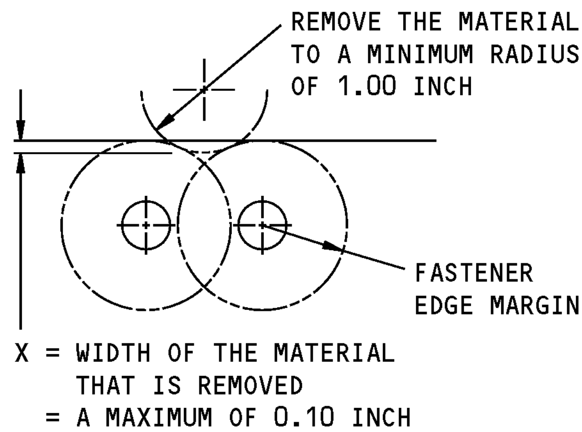



**Cleanup and Sealing of Edge Erosion**
**Figure 106**



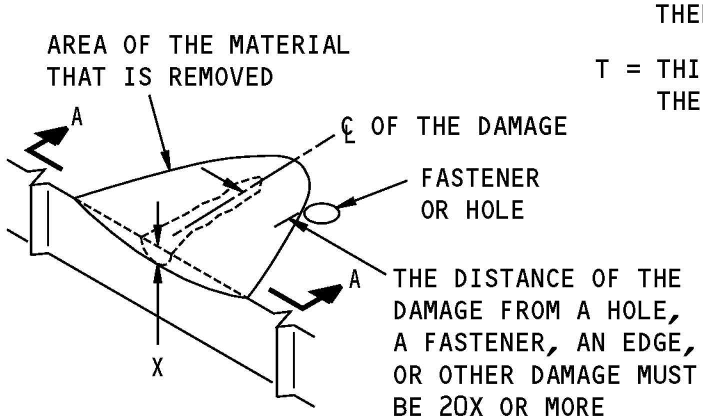

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A



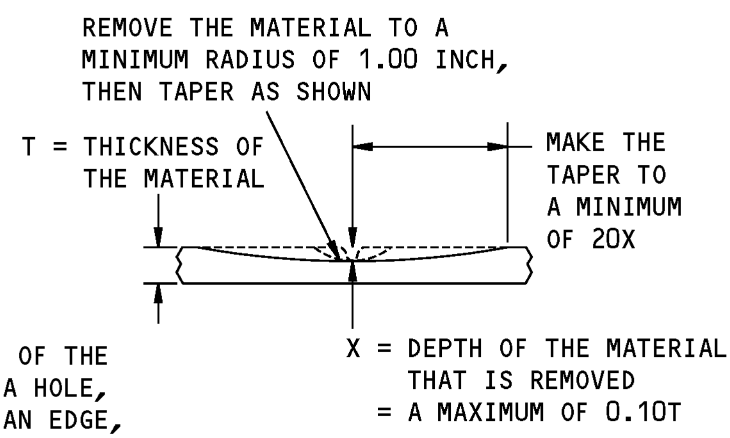

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B



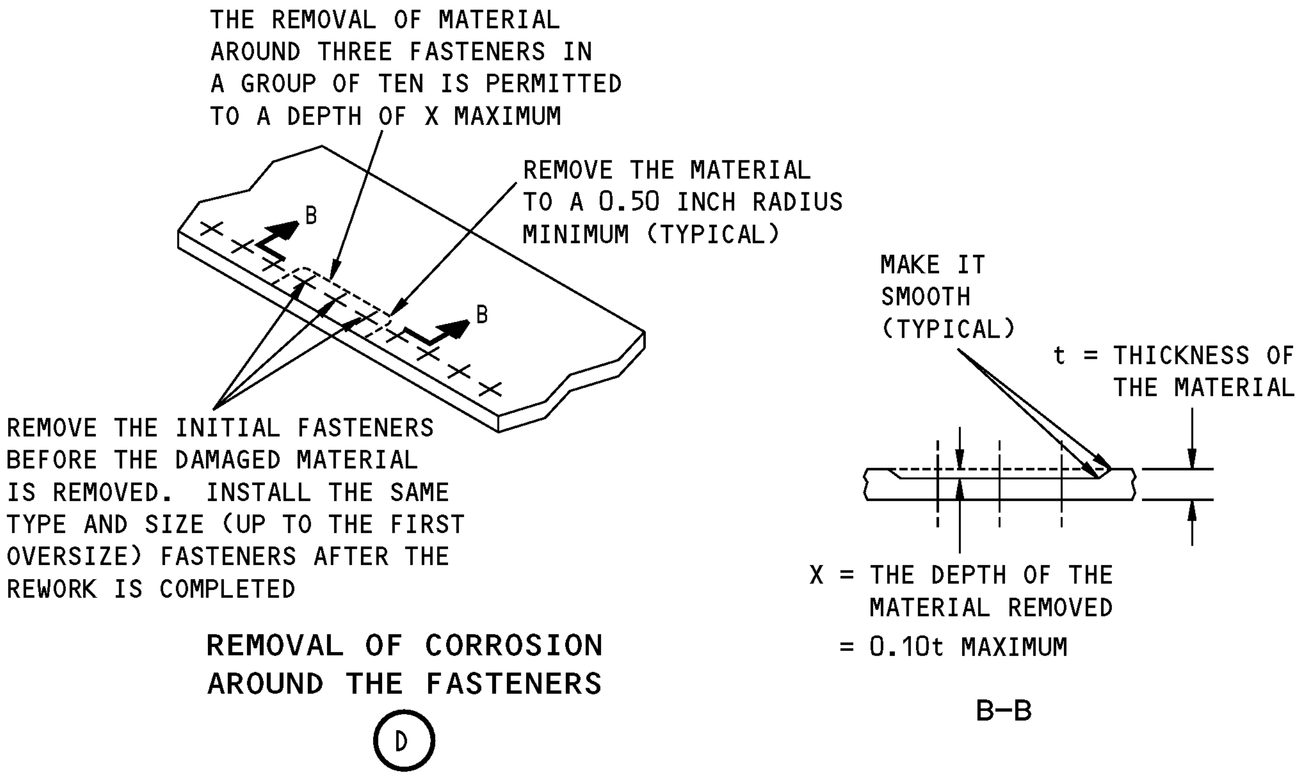

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

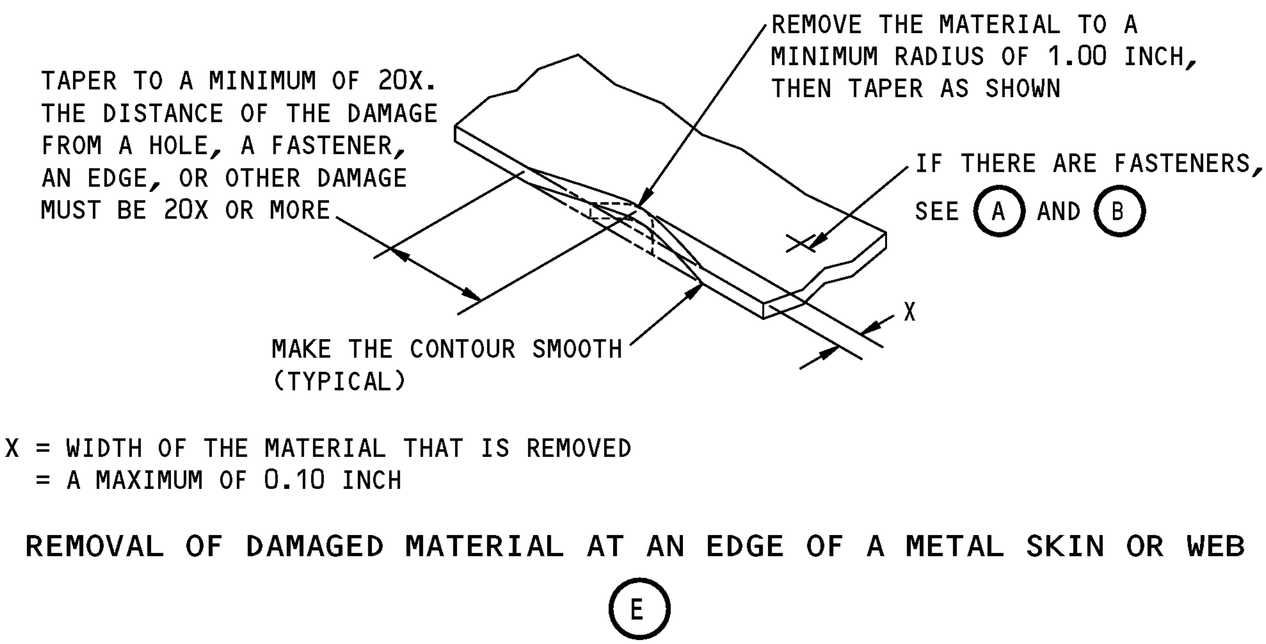
C



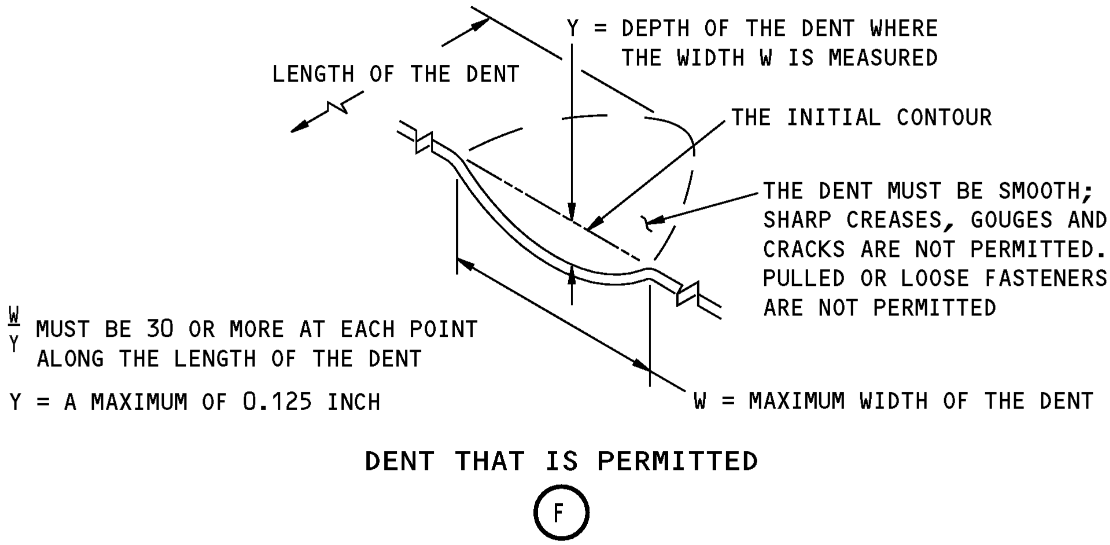

A-A

**Allowable Damage Limits**
**Figure 107 (Sheet 1 of 3)**




**Allowable Damage Limits**
**Figure 107 (Sheet 2 of 3)**



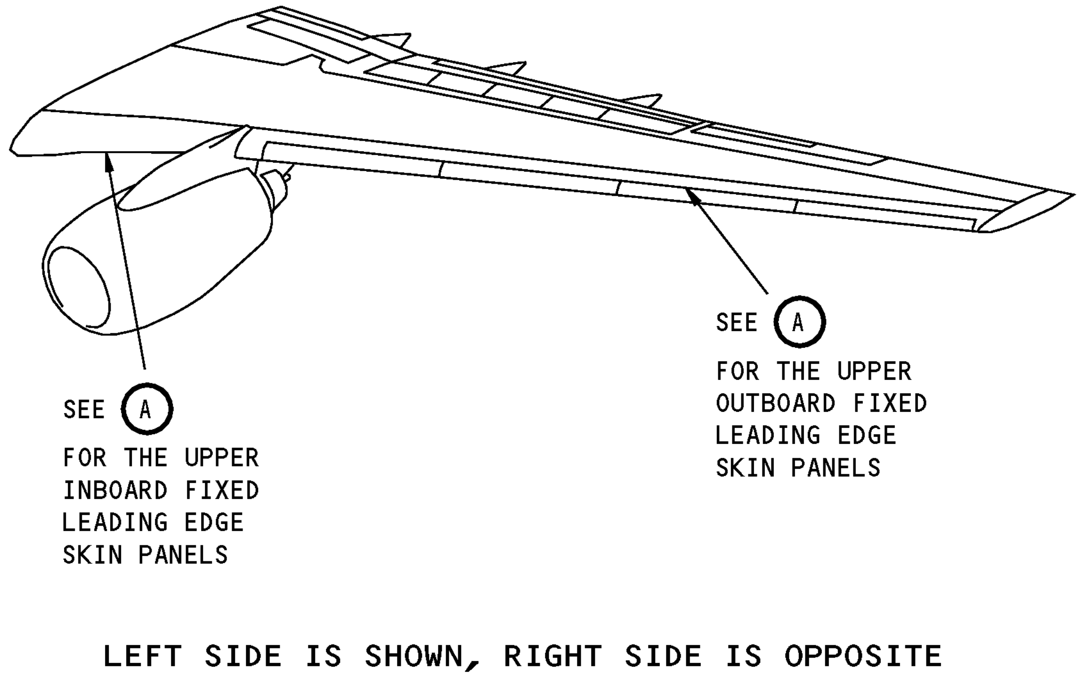

**Allowable Damage Limits**
**Figure 107 (Sheet 3 of 3)**

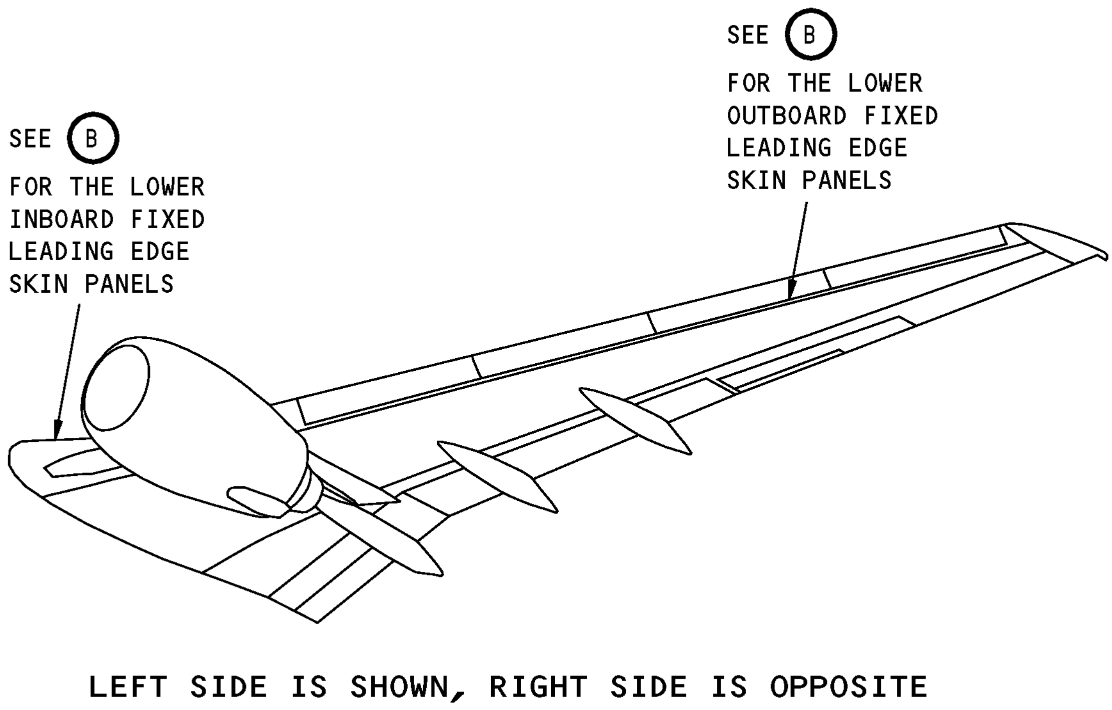
## REPAIR 1 - WING FIXED LEADING EDGE SKIN PANELS MADE OF COMPOSITE MATERIALS

### 1. Applicability

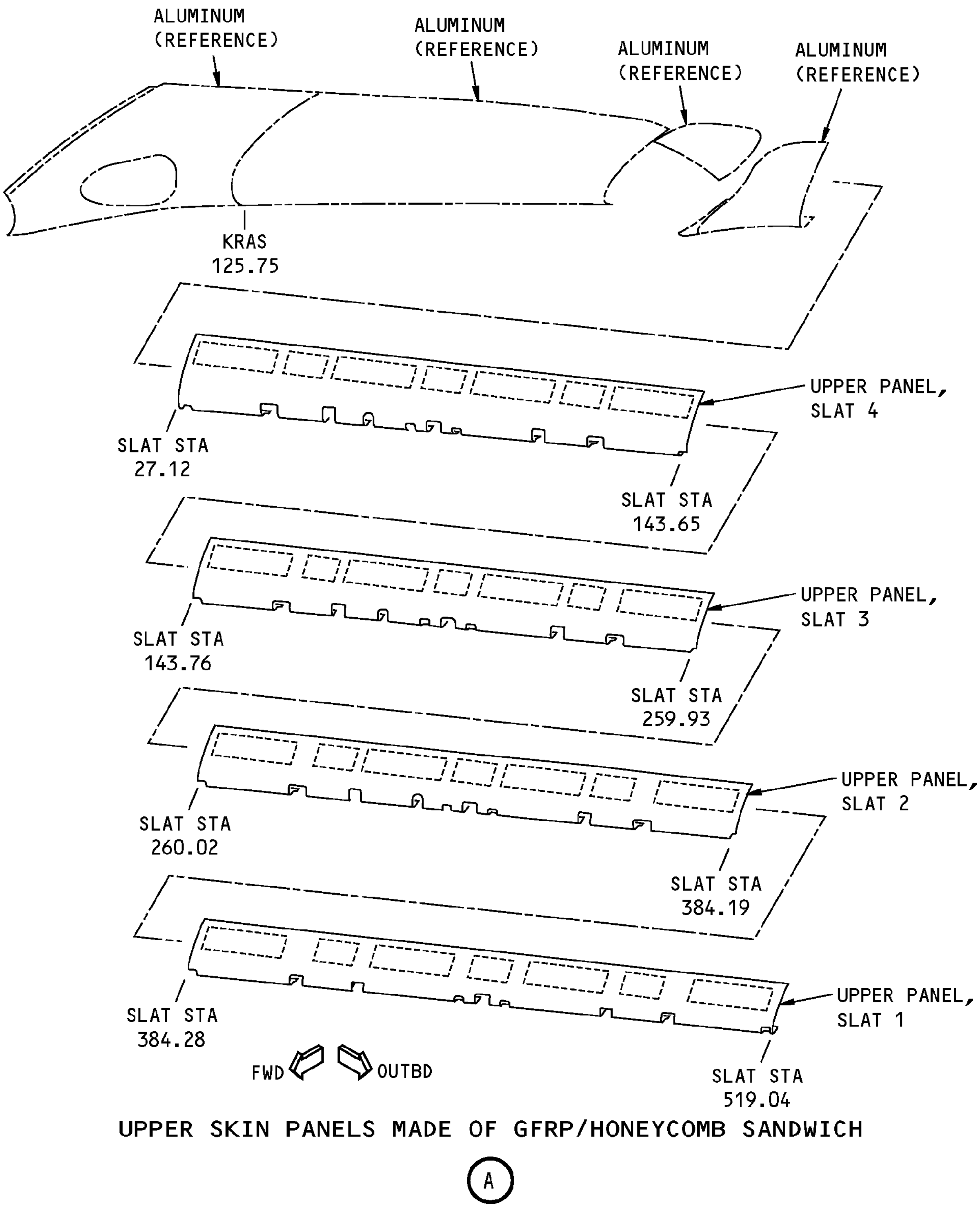
A. Repair 1 is applicable to damage to the fixed leading edge skin panels made of Glass Fiber Reinforced Plastic (GFRP) as shown in Wing Fixed Leading Edge Skin Panel Location, Figure 201/REPAIR 1.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

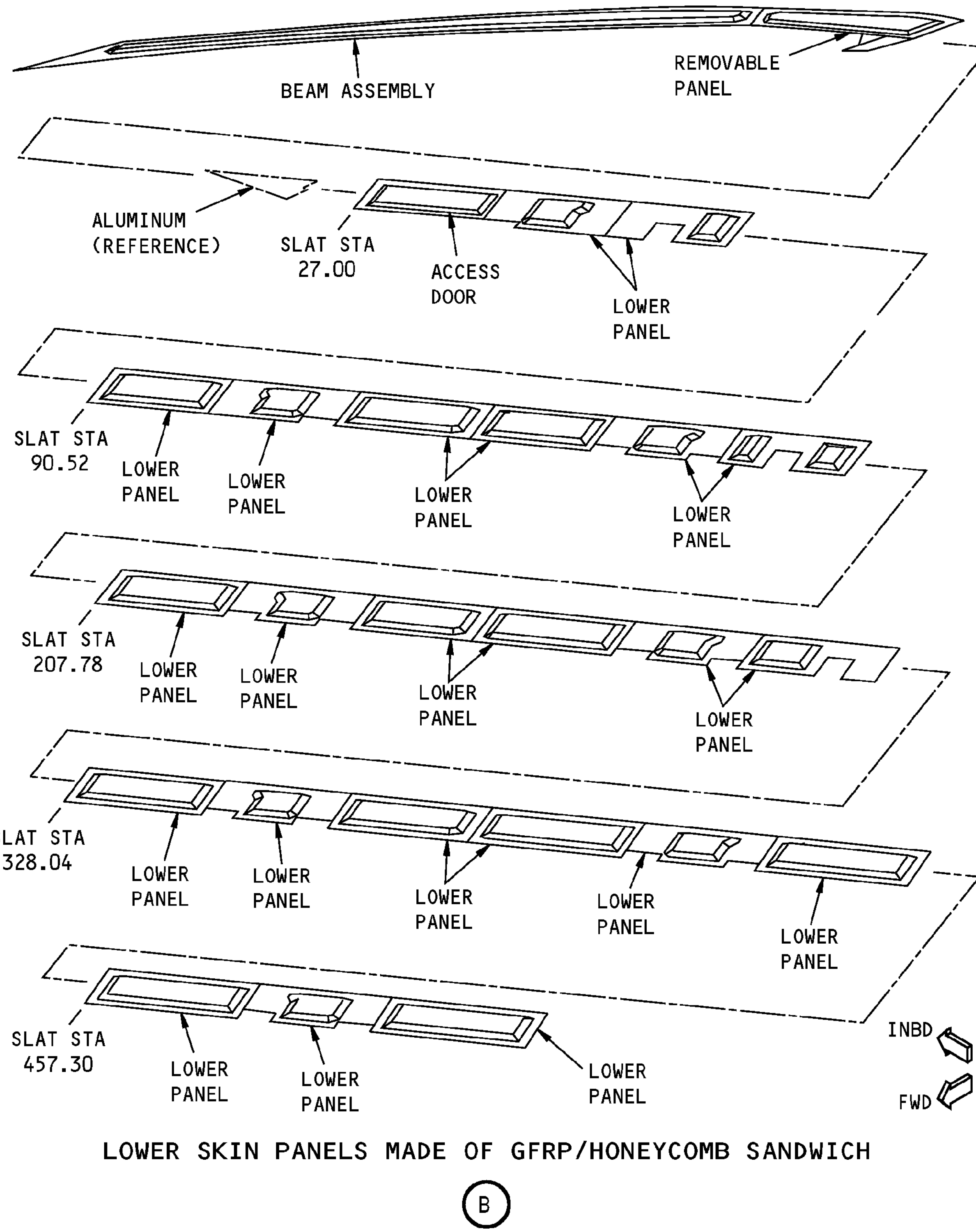



**Wing Fixed Leading Edge Skin Panel Location**
**Figure 201 (Sheet 1 of 3)**




**Wing Fixed Leading Edge Skin Panel Location**
**Figure 201 (Sheet 2 of 3)**




**Wing Fixed Leading Edge Skin Panel Location**
**Figure 201 (Sheet 3 of 3)**

2. **General**

A. Repair 1 gives repair instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the panels as necessary to get access to the inner surface of the skin.

(1) Remove the necessary fasteners. Refer to 51-40-02 for information on fastener removal.

(2) If a fastener hole is damaged, refer to 51-70-04 or 51-70-05, as applicable.

C. Refer to Definitions of the Damage Size, Figure 202/REPAIR 1 for the definitions of the length, width, and depth of damage.

D. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

E. Some GFRP leading edge panels have a thin layer of aluminum on the outer surface.

(1) The aluminum layer can be one of the materials that follow:

(a) BMS 8-278 aluminum coated glass fabric

(b) BMS 8-289 bonded aluminum foil

(c) BMS 8-336 expanded aluminum foil mesh

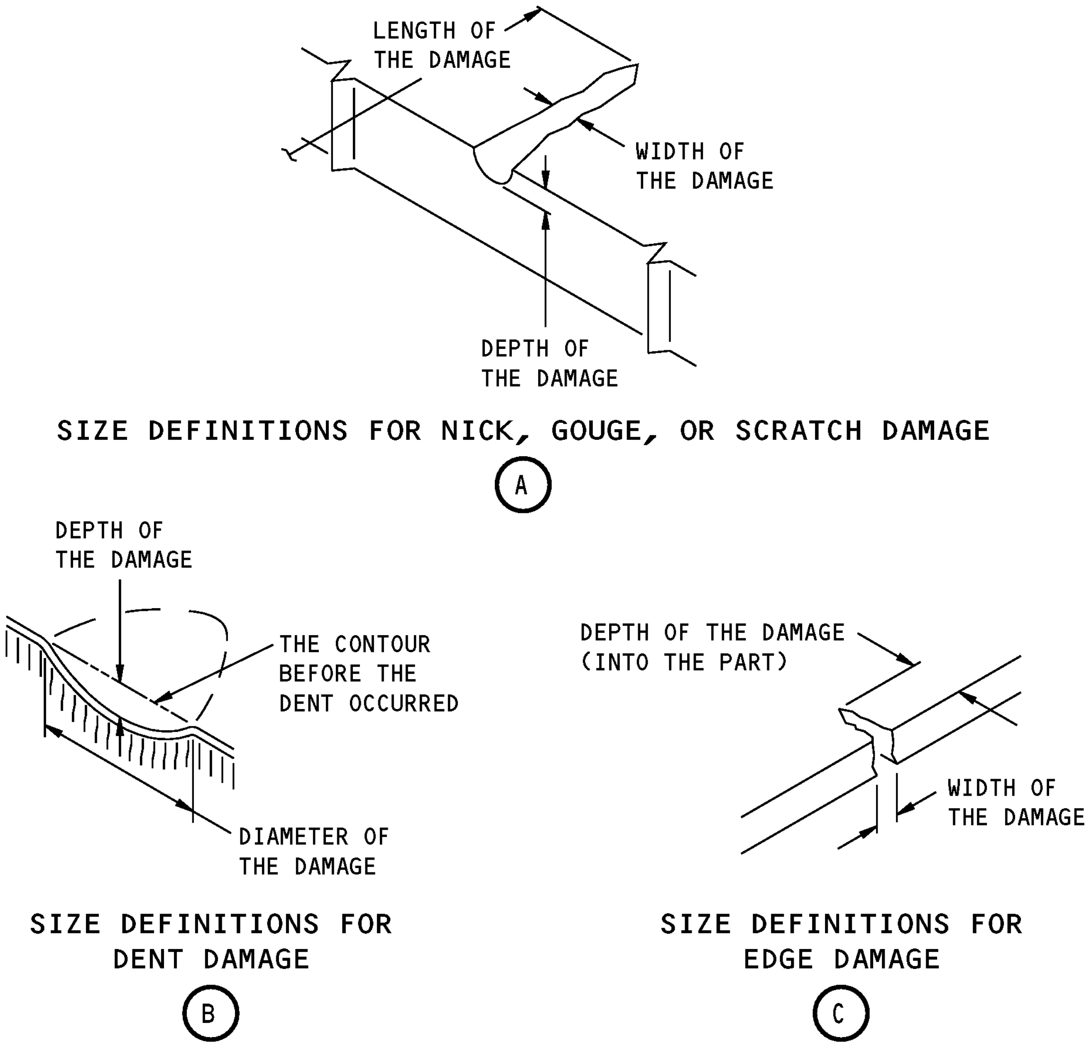
(2) Refer to 51-70-14 before you repair a skin panel that has an aluminum layer.

(3) Refer to 51-70-14 after you repair a skin panel that has an aluminum layer.

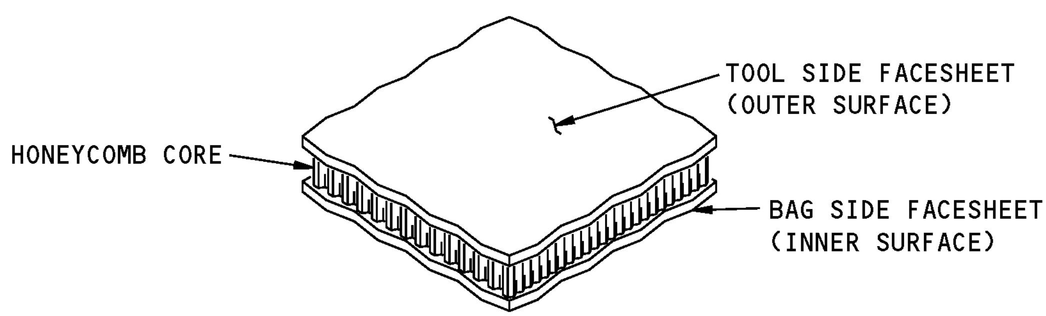
F. Do the repair as given in Paragraph 4./REPAIR 1

G. Install the wing fixed leading edge panels that were removed.

H. Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.




**Definitions of the Damage Size**
**Figure 202**




**Definitions of the Facesheets**
**Figure 203**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-05 | REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS |
| 51-70-14 | STRUCTURES WITH ALUMINUM COATINGS AND FOILS |

## 4. Repair Instructions for the 350°F (177°C) Cure Panels

A. For dents that are a maximum of 2 inches in diameter and have no fiber damage or delamination, do the steps that follow:

(1) Fill the dent with BMS 5-28, Type 7 potting compound

(2) Apply a fiberglass patch over the potted area as given in 51-70-04.

B. For dents that are not permitted by Paragraph 4.A./REPAIR 1 and for other damage that is not permitted by Allowable Damage 1, refer to:

(1) Table 201/REPAIR 1 for panel areas other than the edgebands

(2) Table 202/REPAIR 1 for the edgebands.

**Table 201**:

| REPAIR DATA FOR HONEYCOMB CORE AREAS OF THE 350°F (177°C) CURE WING FIXED LEADING EDGE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | Contact the Boeing Company | Damage that is a maximum of:<br><br>- 6.0 inches in diameter<br><br>One repair for each 144 square inches<br><br>2.0 inches minimum clearance from:<br><br>- other repairs (includes repairs on the opposite panel surface)<br>- fastener holes<br>- damage permitted in Allowable Damage 1<br>- panel edges | Damage that is a maximum of:<br><br>- 6.0 inches in diameter<br><br>One repair for each 144 square inches<br><br>2.0 inches minimum clearance from:<br><br>- other repairs (includes repairs on the opposite panel surface)<br>- fastener holes<br>- damage permitted in Allowable Damage 1<br>- panel edges | There are no limits on the dimensions of the repair |
| REPAIR INSTRUCTIONS | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-05 and Paragraph 4.D | SRM 51-70-05 and Paragraph 4.D |

Table 202:

| REPAIR DATA FOR THE EDGEBANDS OF 350°F (177°C) CURE LEADING EDGE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP REPAIR AS GIVEN IN SRM 51-70-04** | **CATEGORY A WET LAYUP REPAIR AS GIVEN IN SRM 51-70-04** | **CATEGORY A PREIMPREGNATED LAYUP REPAIR AS GIVEN IN SRM 51-70-05** | **CATEGORY A PREIMPREGNATED LAYUP REPAIR AS GIVEN IN SRM 51-70-05** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE | Contact the Boeing Company | Damage that is a maximum of:<br>- 3.0 inches across the largest dimension of the damage<br>- 10 percent of the edgeband length on the side of the damage | There are no limits on the dimensions of the repair | There are no limits on the dimensions of the repair |
| REPAIR INSTRUCTIONS | Refer to Paragraph 4.C | Refer to Paragraph 4.D | Refer to Paragraph 4.E | Refer to Paragraph 4.F |

C. Use the instructions that follow to do a Category B repair with wet layup materials at 150°F (66°C) cure.

(1) Repair the damage as given in 51-70-04, but for each facesheet or solid laminate area that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that were removed.

(b) Do an inspection of the repair at each 800 flight hour interval or more frequently.

1) Replace the repair with a Category A repair if you find deterioration.

D. Use the instructions that follow to do a Category A repair with wet layup materials at 200°F (93°C) cure.

(1) Repair the damage as given in 51-70-04, but for each facesheet or solid laminate area that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that were removed.

(b) Add one structural ply of BMS 9-3, Type H-2, or Type H-3 glass fabric that is ±45 degrees.

(c) Add a second structural ply of BMS 9-3, Type H-2 or Type H-3 glass fabric that is 0 or 90 degrees.

E. Use the instructions that follow to do a Category A repair with preimpregnated layup materials at 250°F (121°C) cure.

(1) Repair the damage as given in 51-70-05, but for each facesheet or solid laminate area that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that were removed.

(b) Add one structural ply of BMS 8-79 glass fabric that is ±45 degrees.

(c) Add a second structural ply of BMS 8-79 glass fabric that is 0 or 90 degrees.

F. Use the instructions that follow to do a Category A repair with preimpregnated layup materials at 350°F (177°C) cure.

(1) Repair the damage as given in 51-70-05.

(2) Use the same number of repair plies as the number of initial plies that were removed.

## REPAIR 2 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE BETWEEN RIBS AND STRINGERS

1. **Applicability**

   A. Repair 2 is applicable to damage:

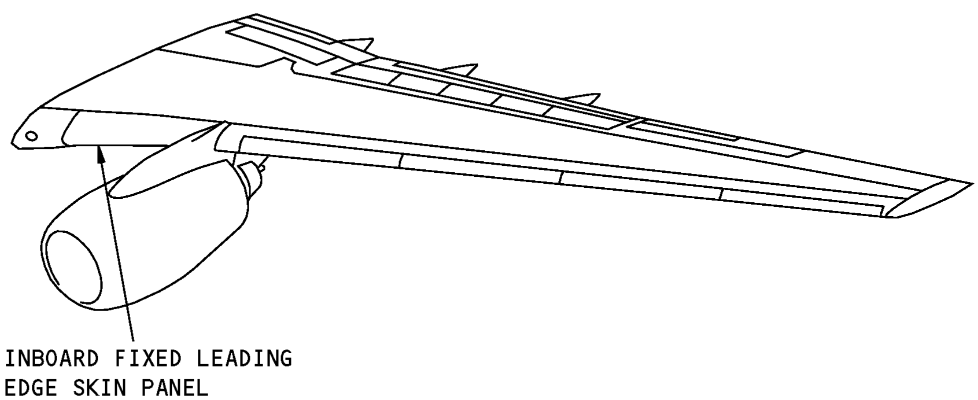
   (1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 2

   (2) That is between ribs and stringers.

2. **General**

   A. Repair 2 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

   B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.




**Wing Fixed Leading Edge Skin Location**
**Figure 201**

3. **References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |

(Continued)

| Reference | Title |
|---|---|
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. **Repair Instructions**

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts.

(1) Make the part [1] doubler and part [2] filler. Refer to Table 201/REPAIR 2.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 2.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Repair Plate | 1 | Use clad or bare 2024-T3 that is 0.071 inch thick |
| [2] | Filler | 1 | Use clad 2024-T3 that is 0.063 inch thick |

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

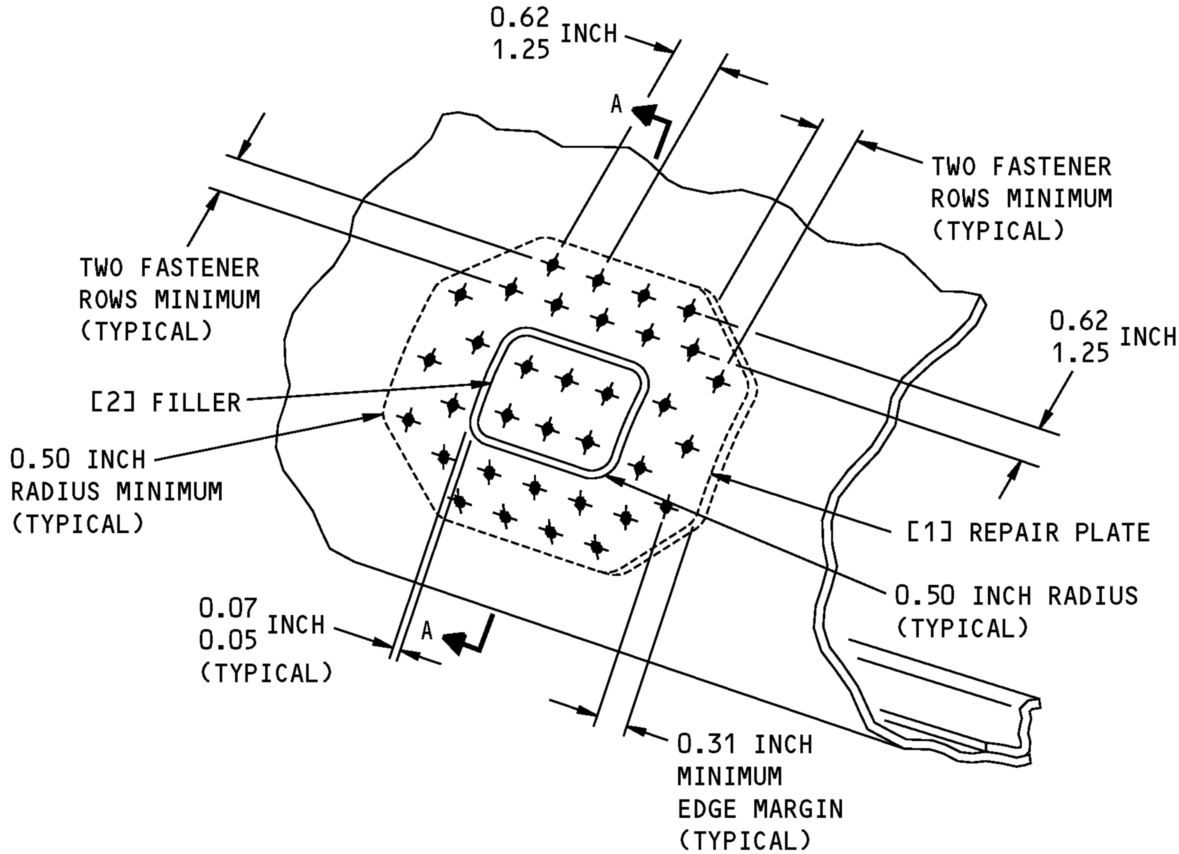
K. Install the repair parts.

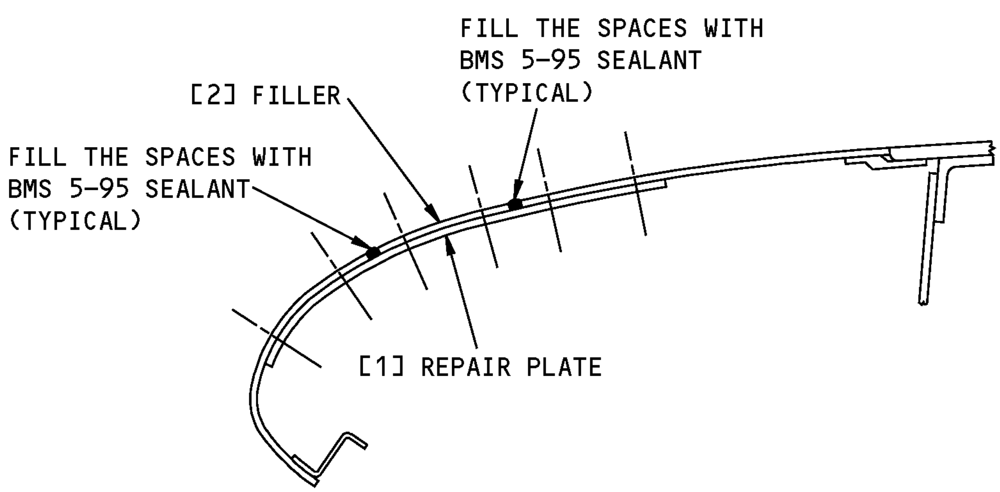
(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [2] filler and the initial skin with BMS 5-95 sealant.




A-A

FASTENER SYMBOLS

⌖ REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW5K HEX DRIVE BOLT AND A BACC30M COLLAR.

**Layout of the Repair Parts**
**Figure 202**

## REPAIR 3 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE AT THE FRONT SPAR

### 1. Applicability

A. Repair 3 is applicable to damage:

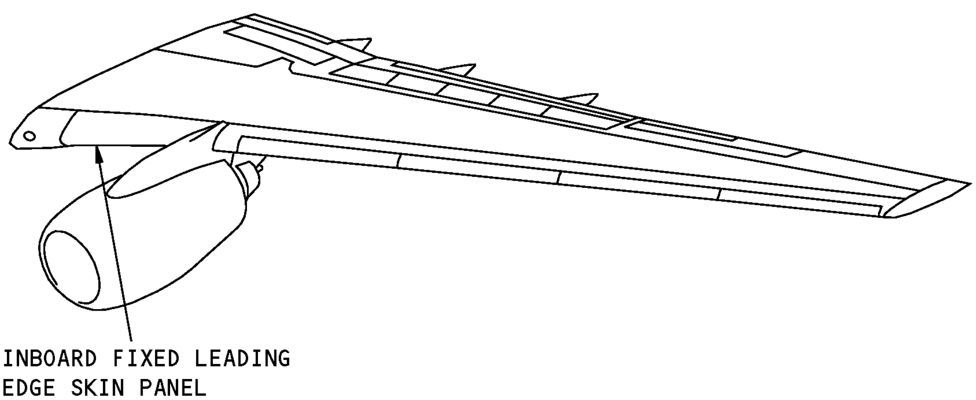
(1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 3

(2) That is at the front spar.

### 2. General

A. Repair 3 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.




**Wing Fixed Leading Edge Skin Location**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |

(Continued)

| Reference | Title |
|---|---|
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. **Repair Instructions**

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts. Refer to Table 201/REPAIR 3.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 3.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Filler | 1 | Use clad 2024-T3 that is 0.063 inch thick |
| [2] | Tapered Filler | 1 | Use clad or bare 2024-T3. Use the same thickness as the attach strap |
| [3] | Doubler | 1 | Use clad or bare 2024-T3 that is 0.071 inch thick |

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

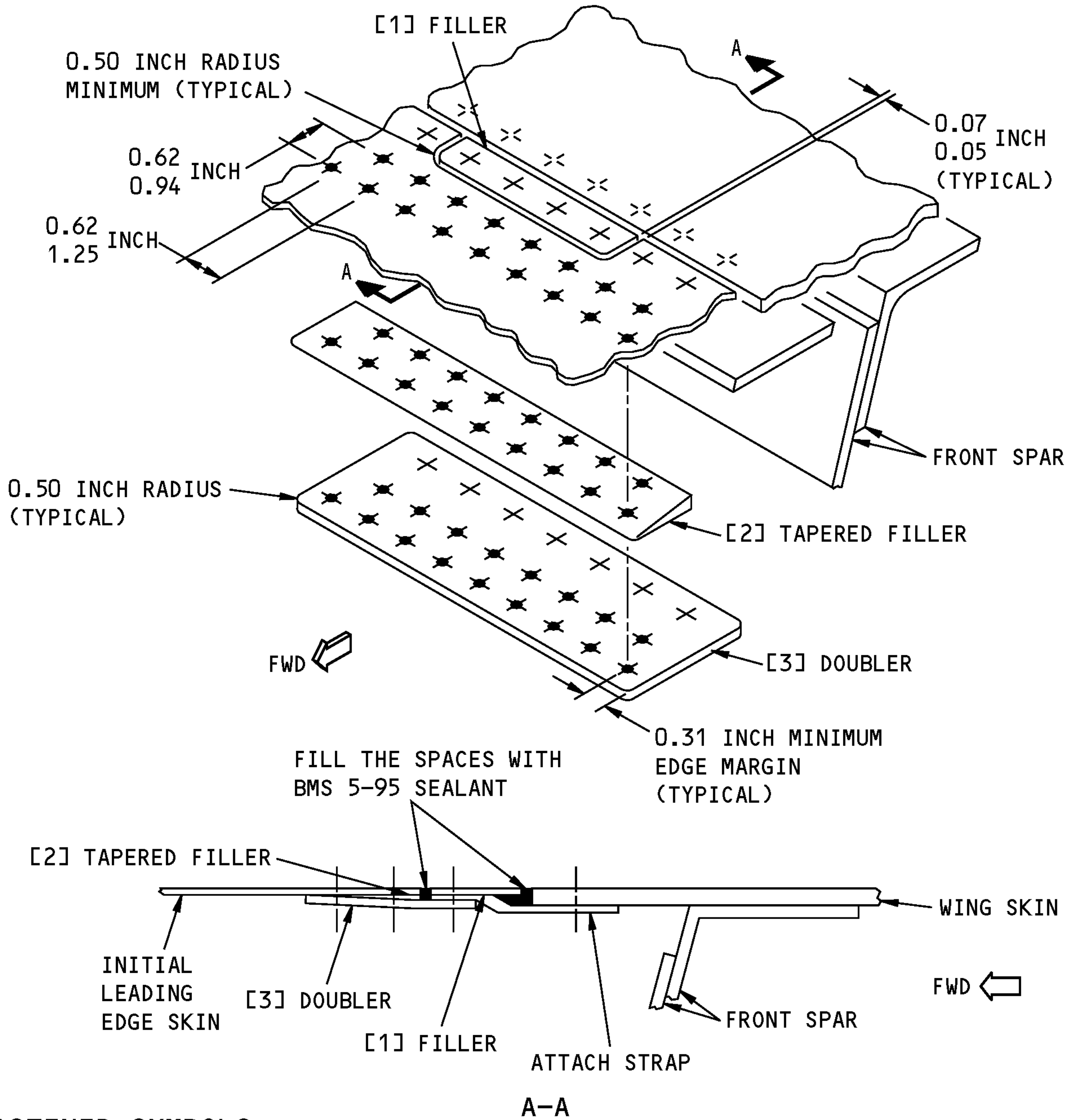
K. Install the repair parts.

(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.




FASTENER SYMBOLS

- REFERENCE FASTENER LOCATION.
- INITIAL FASTENER LOCATION. INSTALL A FASTENER THAT IS THE SAME TYPE AND DIAMETER AS THE INITIAL FASTENER. YOU CAN USE A FASTENER THAT IS UP TO 1/32 INCH DIAMETER OVERSIZE.
- REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW(5)K HEX DRIVE BOLT AND A BACC30M COLLAR.

**Layout of the Repair Parts**
**Figure 202**

## REPAIR 4 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE AT A RIB

### 1. Applicability

A. Repair 4 is applicable to damage:

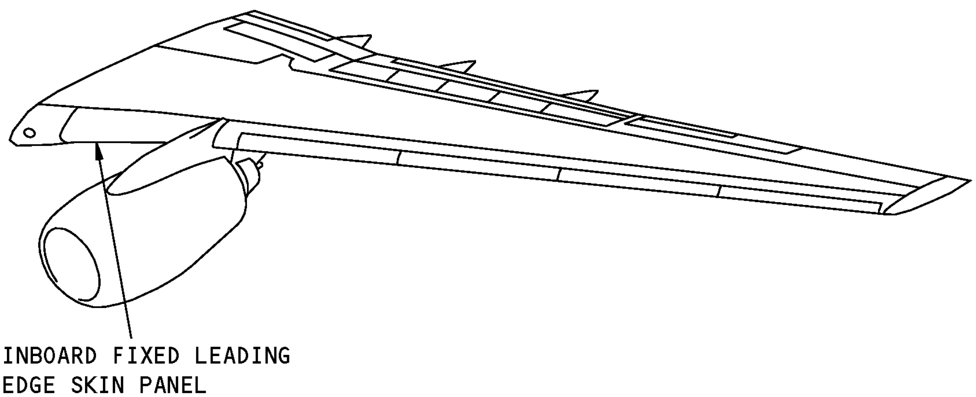
(1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 4

(2) That is at a rib.

### 2. General

A. Repair 4 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.




**Wing Fixed Leading Edge Skin Location**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |

(Continued)

| Reference | Title |
|---|---|
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Repair Instructions

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts. Refer to Table 201/REPAIR 4.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 4.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Filler | 1 | Use clad 2024-T3 that is 0.063 inch thick |
| [2] | Doubler | 2 | Use clad or bare 2024-T3 that is 0.071 inch thick |
| [3] | Filler | 2 | Use clad or bare 2024-T3 that is 0.080 inch thick |

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

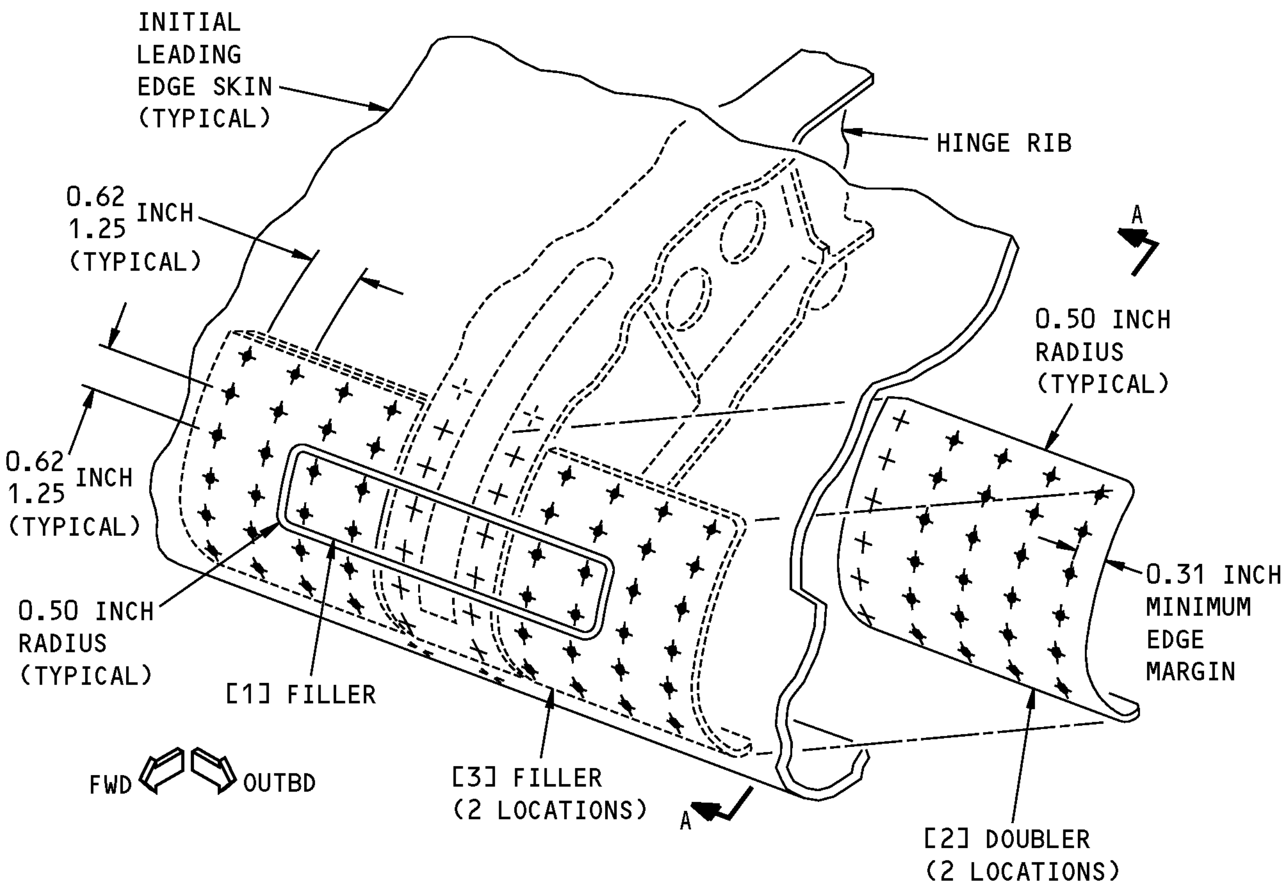
K. Install the repair parts.

(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.



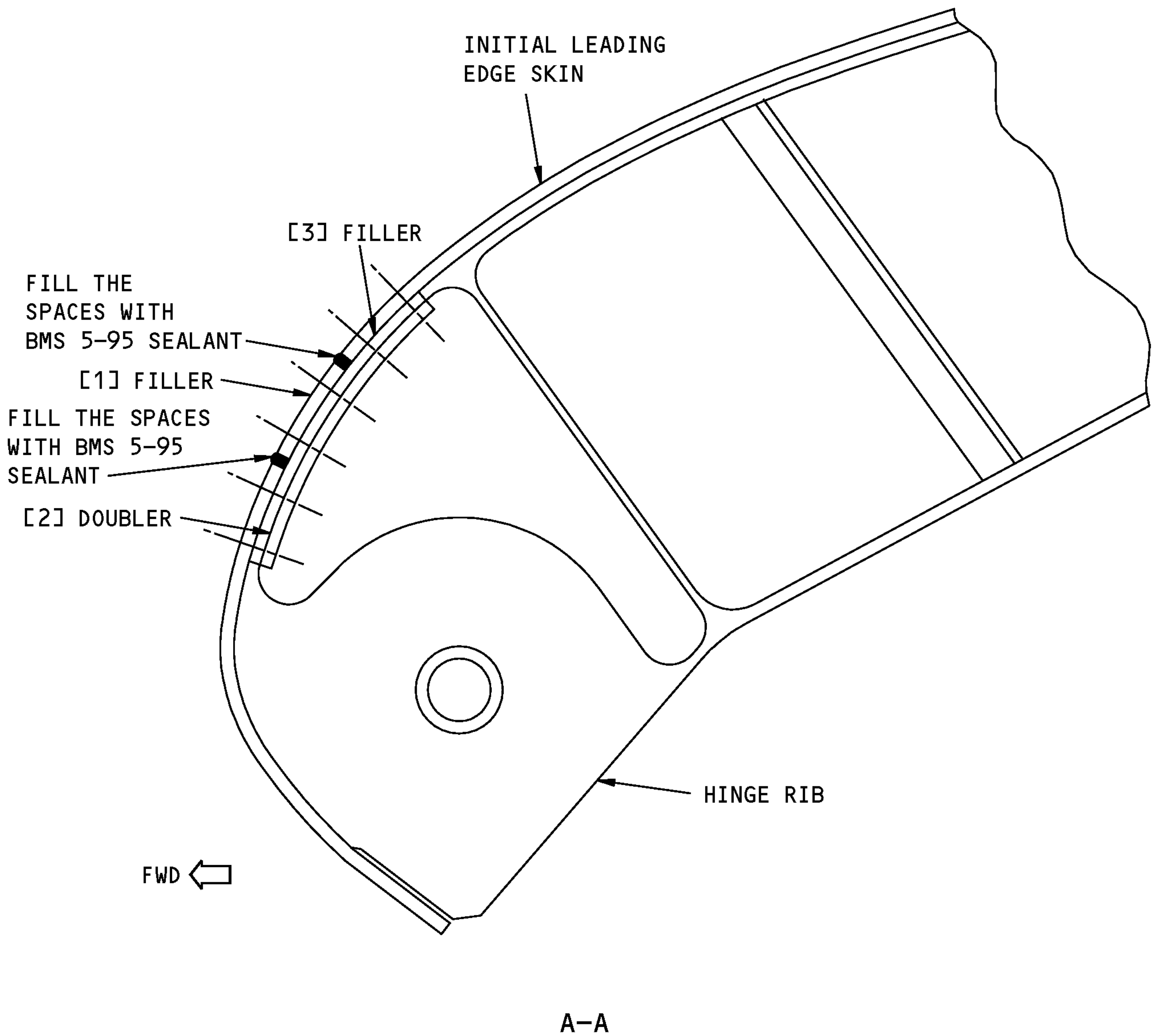

FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

+ INITIAL FASTENER LOCATION. INSTALL A FASTENER THAT IS THE SAME TYPE AND DIAMETER AS THE INITIAL FASTENER. YOU CAN USE A FASTENER THAT IS UP TO 1/32 INCH DIAMETER OVERSIZE.

REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW5K HEX DRIVE BOLT AND A BACC30M COLLAR.

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

## REPAIR 5 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE AT A STRINGER

### 1. Applicability

A. Repair 5 is applicable to damage:

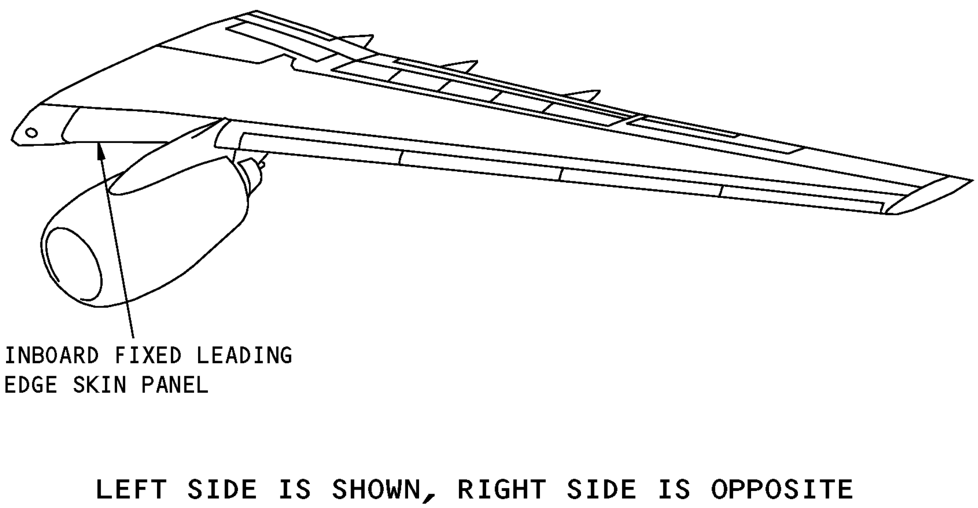
(1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 5

(2) That is at a stringer.

### 2. General

A. Repair 5 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.




**Wing Fixed Leading Edge Skin Location**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |

(Continued)

| Reference | Title |
|---|---|
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. **Repair Instructions**

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts Refer to Table 201/REPAIR 5.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 5.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Filler | 1 | Use clad 2024-T3 that is 0.063 inch thick |
| [2] | Doubler | 1 | Use clad or bare 2024-T3 that is 0.071 inch thick |
| [3] | Filler | 1 | Use clad or bare 2024-T3 that is 0.070 inch thick. |
| [4] | Angle | 1 | Use clad or bare 2024 in the annealed condition that is 0.071 inch thick. Heat treat to the T42 condition after you bend the part |

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

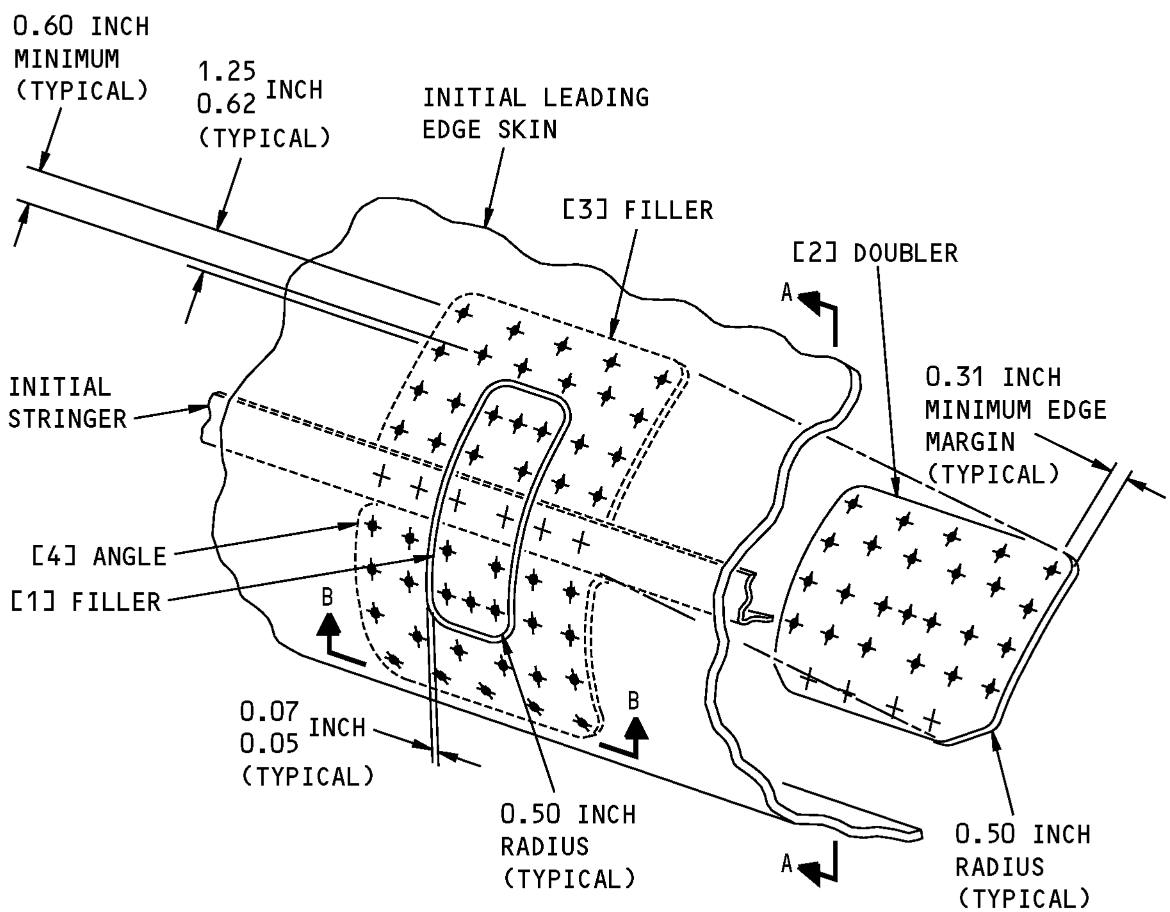
K. Install the repair parts.

(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.



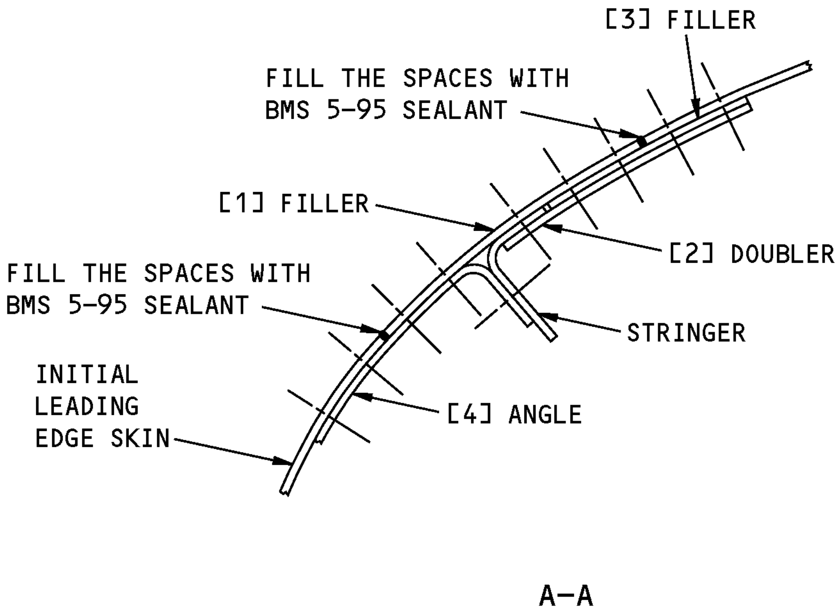

FASTENER SYMBOLS

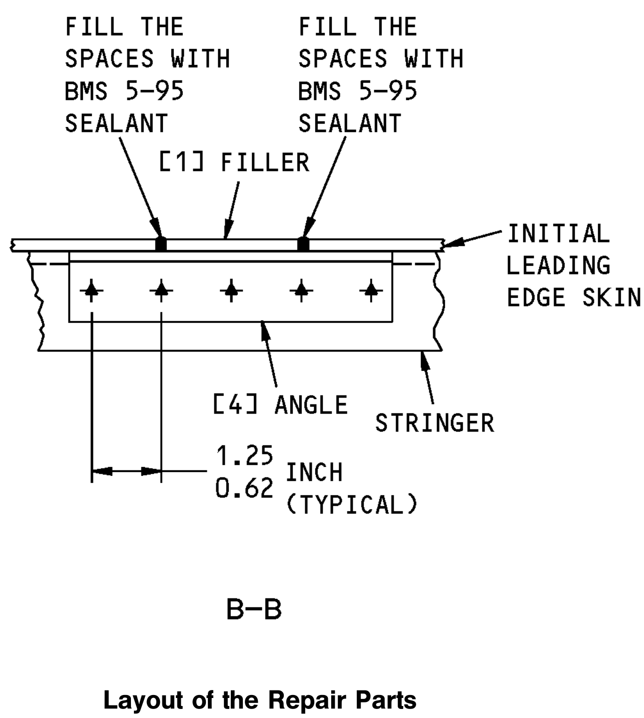
+ INITIAL FASTENER LOCATION. INSTALL A FASTENER THAT IS THE SAME TYPE AND DIAMETER AS THE INITIAL FASTENER. YOU CAN USE A FASTENER THAT IS UP TO 1/32 INCH DIAMETER OVERSIZE.

REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW5K HEX DRIVE BOLT AND A BACC30M COLLAR.

REPAIR FASTENER LOCATION. INSTALL A BACR15BB6D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30MY5K HEX DRIVE BOLT AND A BACC30M COLLAR.

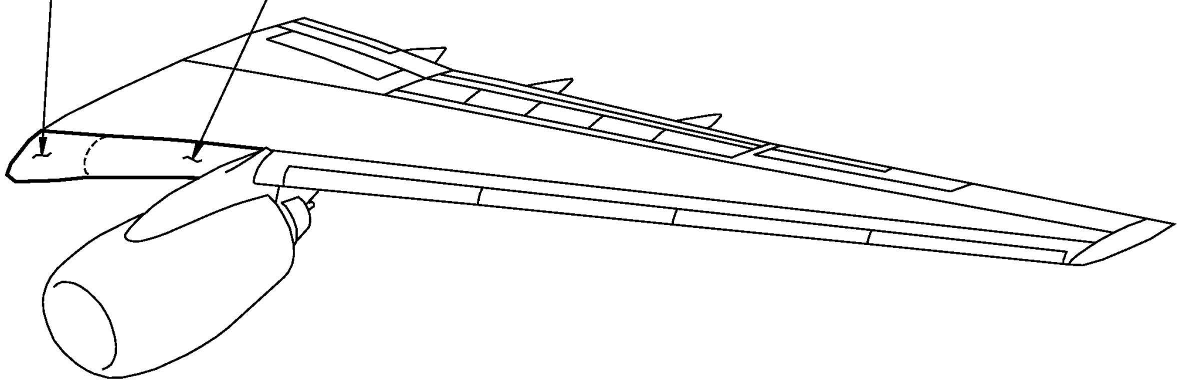
**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

## IDENTIFICATION 1 - WING INBOARD FIXED LEADING EDGE STRUCTURE



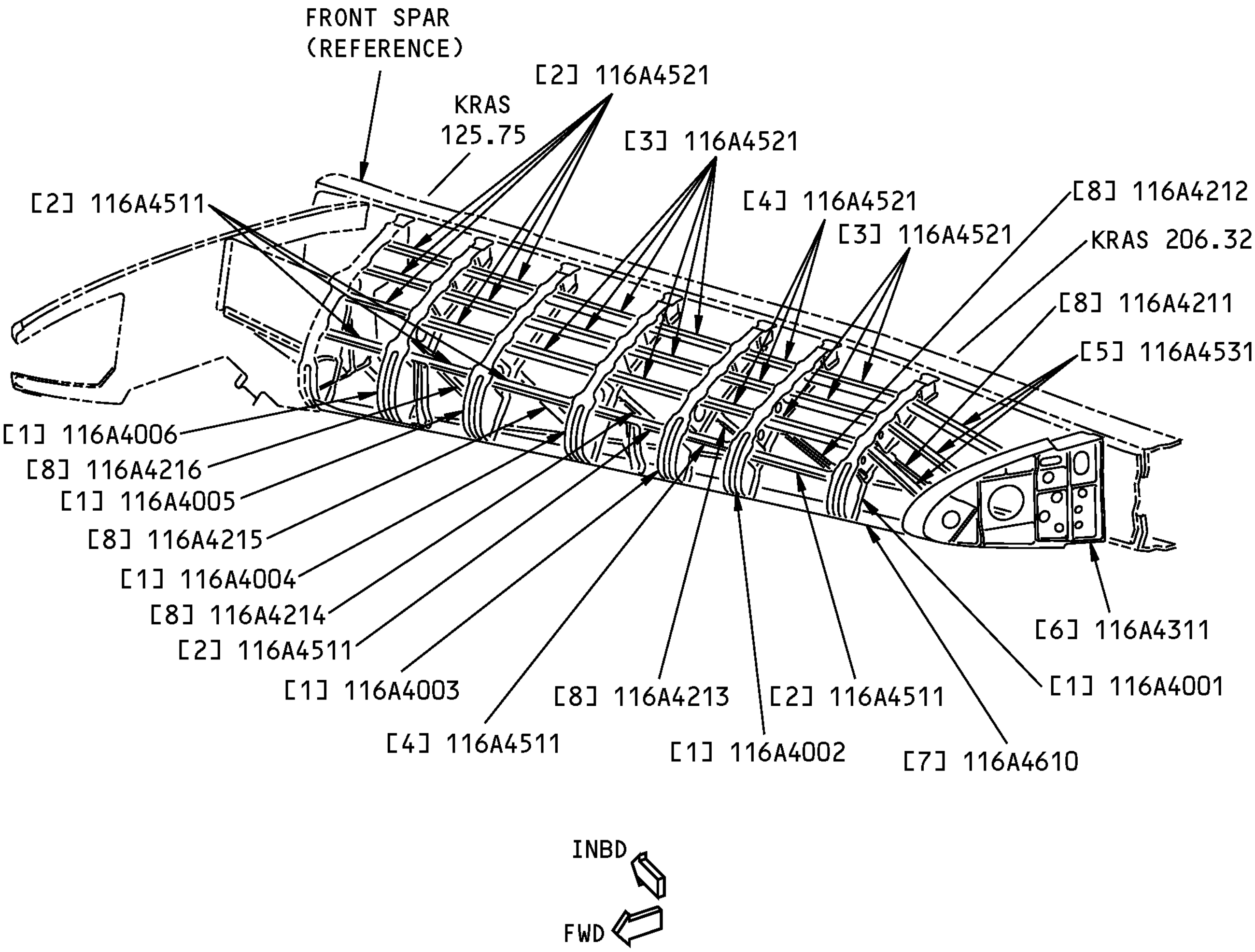

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Inboard Fixed Leading Edge Structure Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A1001 | Wing Left - Product Collector |
| 116A0010 | Fixed Leading Edge Collector - Final Assembly |
| 116A0015 | Functional Product Collector - Front Spar Leading Edge Join, Left Wing |
| 116A3001 | Strakelet Installation - Inboard Fixed Leading Edge Wing |
| 116A3010 | Strakelet Assembly - Inboard Fixed Leading Edge Wing |
| 116A4008 | Skin Installation - Upper, Inboard leading Edge, Wing |



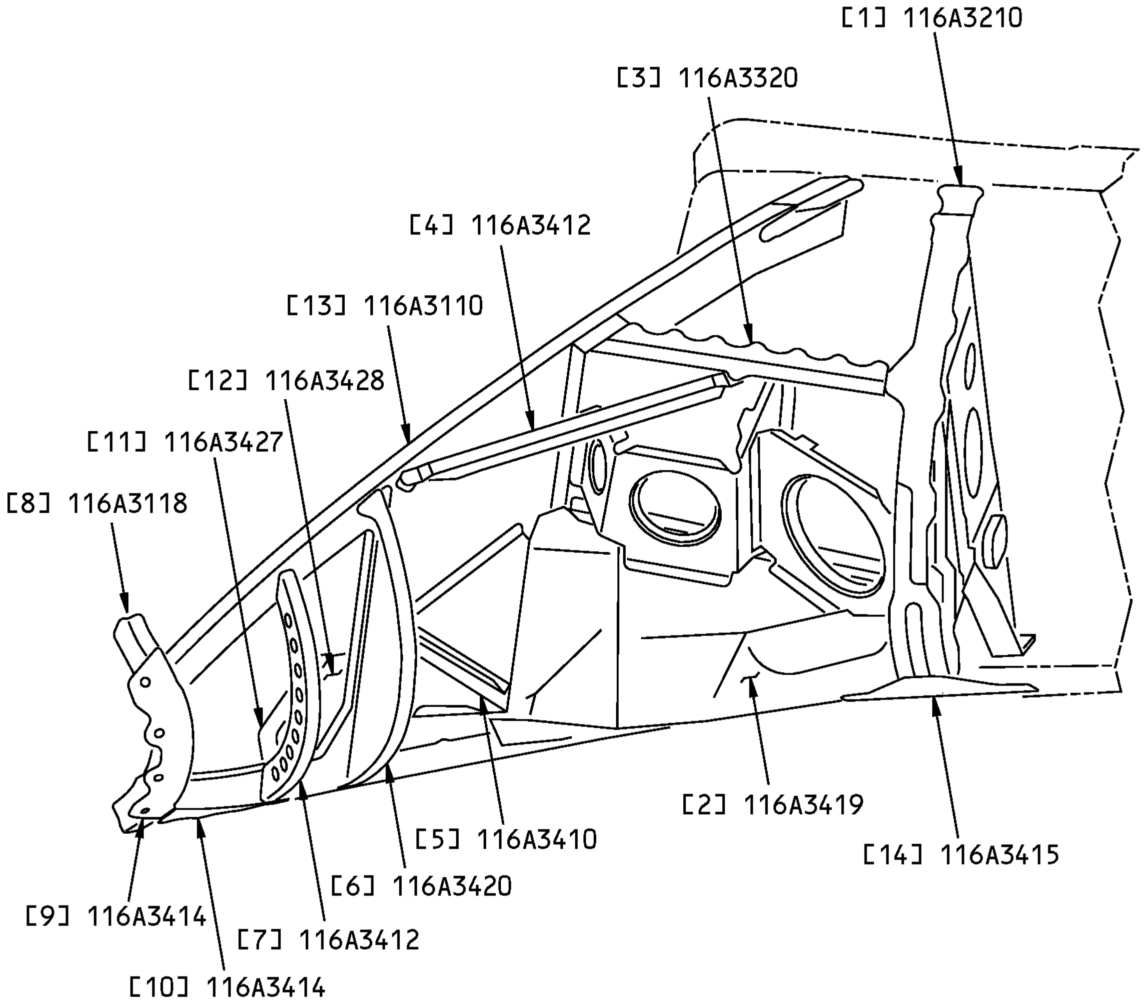

LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

**Wing Inboard Fixed Leading Edge Structure**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Hinge Rib (6) | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part | |
| [2] | Stringer (11) | 0.071 (1.8) | 7075-T62 sheet as given in in QQ-A-250/12 | |
| [3] | Stringer (9) | 0.080 (2.03) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [4] | Stringer (4) | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [5] | Stringer (3) | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part | |
| [6] | Seal Rib | | 7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part | |
| [7] | Nose Beam | 0.063 (1.6) | 7075-T62 clad sheet as given in BMS 7-302 | |
| [8] | Strut | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Wing Inboard Leading Edge Straklet Structure Identification**
**Figure 3**

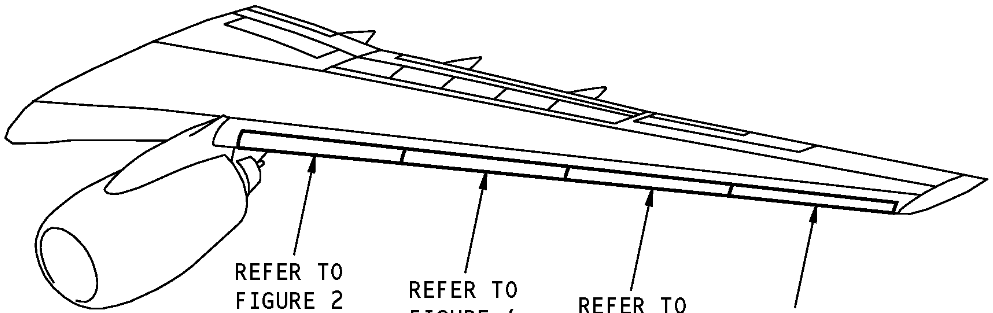
**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Seal Rib Assembly | | | |
| | Seal Rib | | 7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part | |
| | Web | 0.063 (1.6) | 2024-T3 clad sheet as given in QQ-A-250/5 | |
| [2] | Web | 0.063 (1.6) | Ti-6Al-4V titanium sheet as given in MIL-T-9046, Code AB-1, Condition A | |
| [3] | Bulkhead Assembly | | | |
| | Web | 0.032 (0.81) | 2024-T3 clad sheet as given in QQ-A-250/5 | |
| | Chord | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part | |
| | Stiffener (2) | 0.050 (1.27) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| | Stiffener | 0.063 (1.6) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| | Angle | 0.050 (1.27) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [4] | Stiffener | 0.071 (1.8) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [5] | Web Assembly | | | |
| | Web | 0.032 (0.81) | 2024-T42 clad sheet as given in QQ-A-250/5 | |
| | Angle | 0.040 (1.01) | 2024-T42 clad sheet as given in QQ-A-250/5 | |
| [6] | Rib Assembly | | | |
| | Internal Rib | 0.050 (1.27) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| | Angle | 0.050 (1.27) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [7] | Stiffener | 0.050 (1.27 | 7075-T62 sheet as given in QQ-A-250/12 | |
| [8] | Frame | 0.063 (1.01) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [9] | Splice Strap | 0.071 (1.8) | 7075-T6 clad sheet as given in QQ-A-250/13 | |
| [10] | Lower Splice Strap | 0.071 (1.8) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [11] | End Cap | 0.063 (1.6) | 6061-T62 sheet as given in QQ-A-250/11 | |
| [12] | Top Cover Assembly | | | |
| | Stiffener | 0.050 (1.27) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| | Top Cover | 0.040 (1.01) | 7075-T6 clad sheet as given in QQ-A-250/13 | |
| [13] | End Rib Assembly | | | |
| | End Rib | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part | |
| | Attach Angle | 0.050 (1.27) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| | Frame | 0.063 (1.6) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| | Upper Splice Strap | 0.071 (1.80) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [14] | Nose Beam Splice | 0.090 (2.28) | 7075-T62 clad sheet as given in BMS 7-302 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - WING OUTBOARD FIXED LEADING EDGE STRUCTURE

### 1. Locator




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
(EXCEPT AS NOTED)

**Wing Outboard Fixed Leading Edge Structure Locations**
**Figure 1**

### 2. Reference Drawings

A. Reference Drawings

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A1001 | Wing Left - Product Collector |
| 116A0010 | Fixed Leading Edge Collector - Final Assembly |
| 116A0011 | Fixed Leading Edge Collector - Post Paint - Left Wing |
| 116A0013 | Collector - Leading Edge, Wing Build-Up, Left Wing |
| 116A0015 | Functional Product Collector - Front Spar Leading Edge Join, Left Wing |
| 116A5600 | Nose Beam Installation - Outboard Fixed Leading Edge |
| 116A7000 | Rib Installation - Outboard Fixed Leading Edge |

### 3. Applicability

A. Identification 2 is applicable to:

(1) Airplanes that have not had winglets installed.

(2) Airplanes that have had winglets installed as part of the production of the airplane.

**NOTE**: Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

## 4. Wing Outboard Fixed Leading Edge Structure Identification

A. Lists of materials:

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [2] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [3] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [4] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [5] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [6] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [7] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [8] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [9] | Splice Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [10] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [11] | Nose Beam (4) | 0.050 (12.7) | 7075-T62 clad sheet as given in BMS 7-302 | |
| [12] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Splice Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [2] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [3] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [4] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [5] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [6] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [7] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [8] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [9] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [10] | Nose Beam (4) | 0.050 (12.7) | 7075-T62 clad sheet as given in BMS 7-302 | |
| [11] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [12] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [13] | Nose Beam | 0.063 (1.6) | 7075-T62 clad sheet as given in BMS 7-302 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

**Table 4**:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [2] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [3] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [4] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [5] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [6] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [7] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [8] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [9] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [10] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [11] | Splice Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [12] | Nose Beam (4) | 0.050 (12.7) | 7075-T62 clad sheet as given in BMS 7-302 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

**Table 5**:

| LIST OF MATERIALS FOR FIGURE 5 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [2] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [3] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [4] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [5] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [6] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [7] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [8] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [9] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [10] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [11] | Splice Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [12] | Nose Beam (4) | 0.050 (12.7) | 7075-T62 clad sheet as given in BMS 7-302 | |

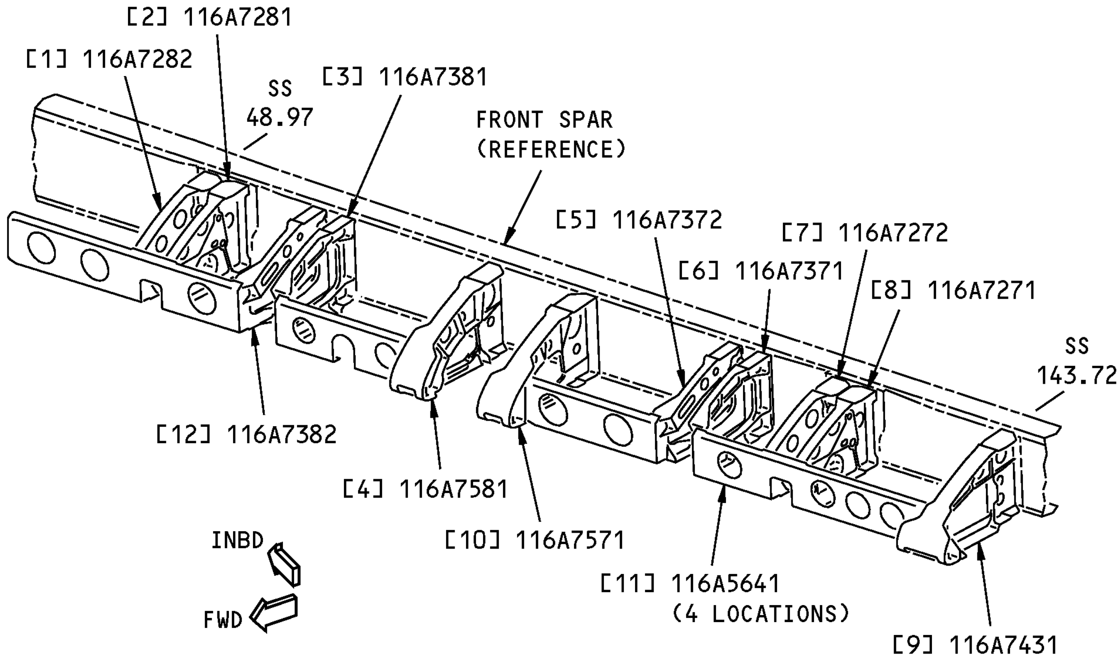
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

**Table 6**:

| LIST OF MATERIALS FOR FIGURE 6 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [2] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [3] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |

| LIST OF MATERIALS FOR FIGURE 6 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [4] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [5] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [6] | Stop Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [7] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [8] | Auxiliary Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [9] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [10] | Main Track Rib | | 7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part) | |
| [11] | Seal Rib | | 7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part) | |
| [12] | Nose Beam (4) | 0.050 (12.7) | 7075-T62 clad sheet as given in BMS 7-302 | |

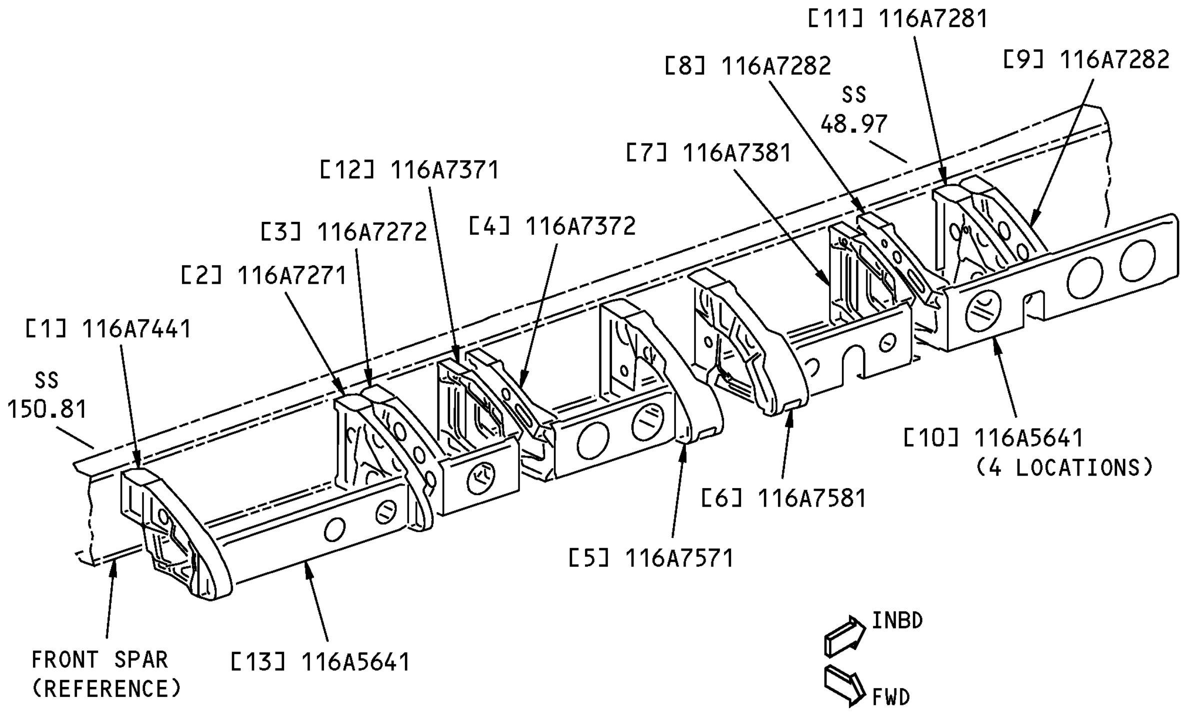
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

SLAT NUMBER 4

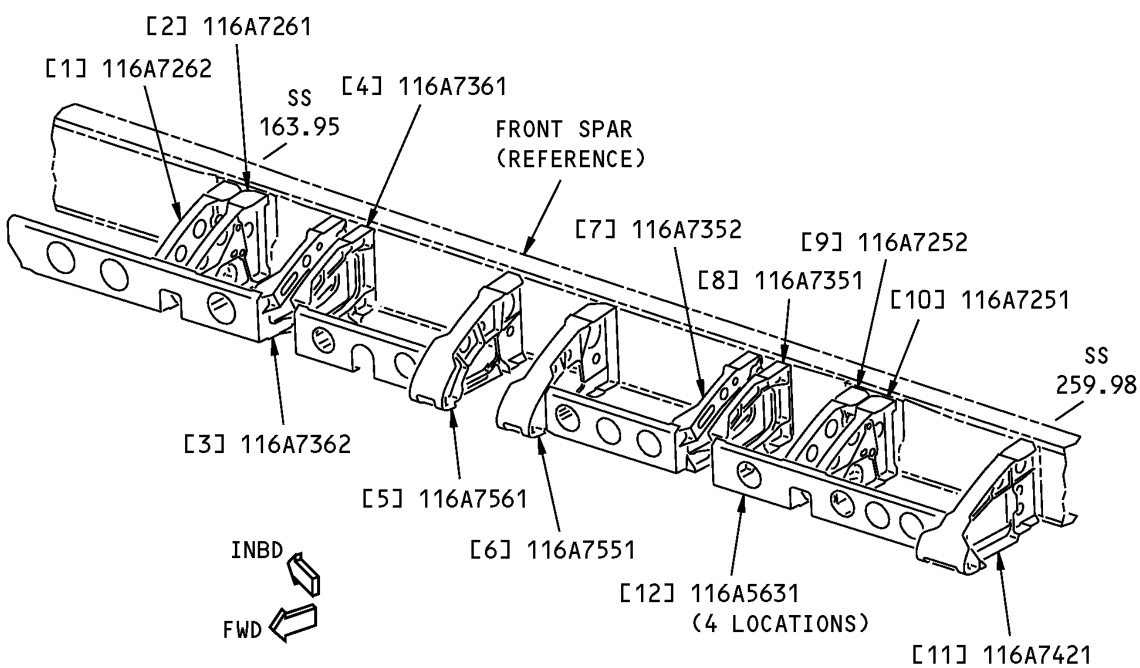
**Wing Outboard Fixed Leading Edge Slat Number 4 Structure Identification**
**Figure 2**




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

SLAT NUMBER 5

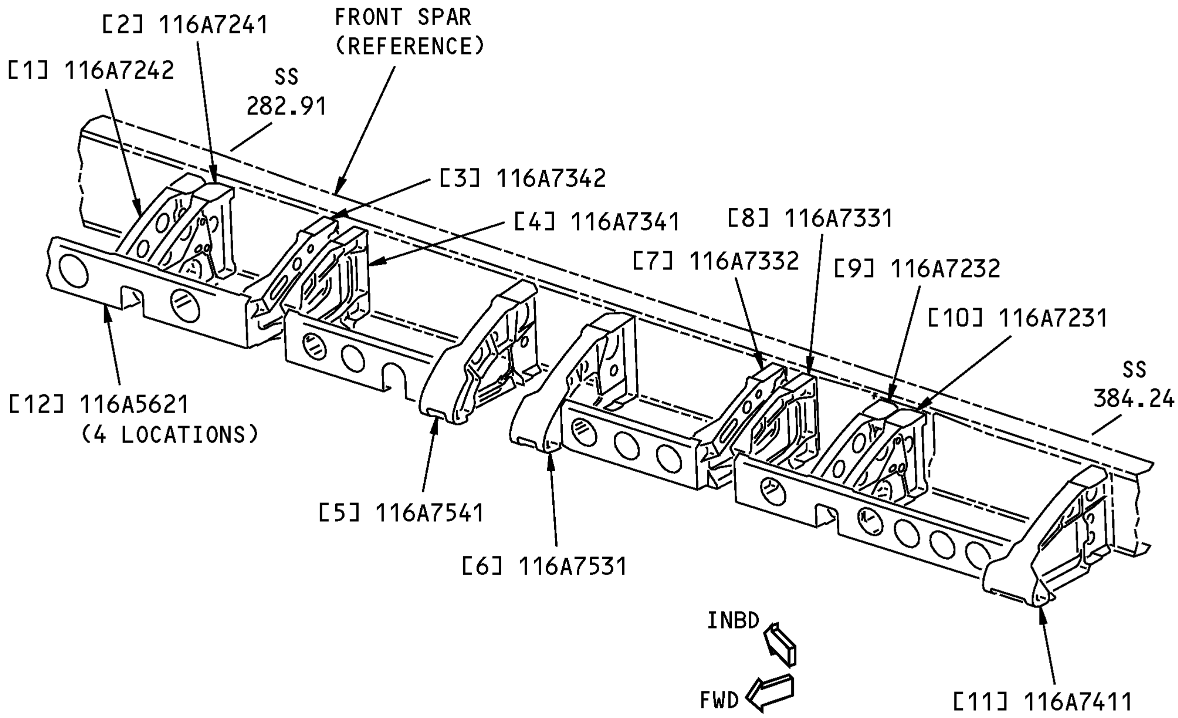
**Wing Outboard Fixed Leading Edge Slat Number 5 Structure Identification**
**Figure 3**




NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

SLAT NUMBER 3 IS SHOWN, SLAT NUMBER 6 IS OPPOSITE

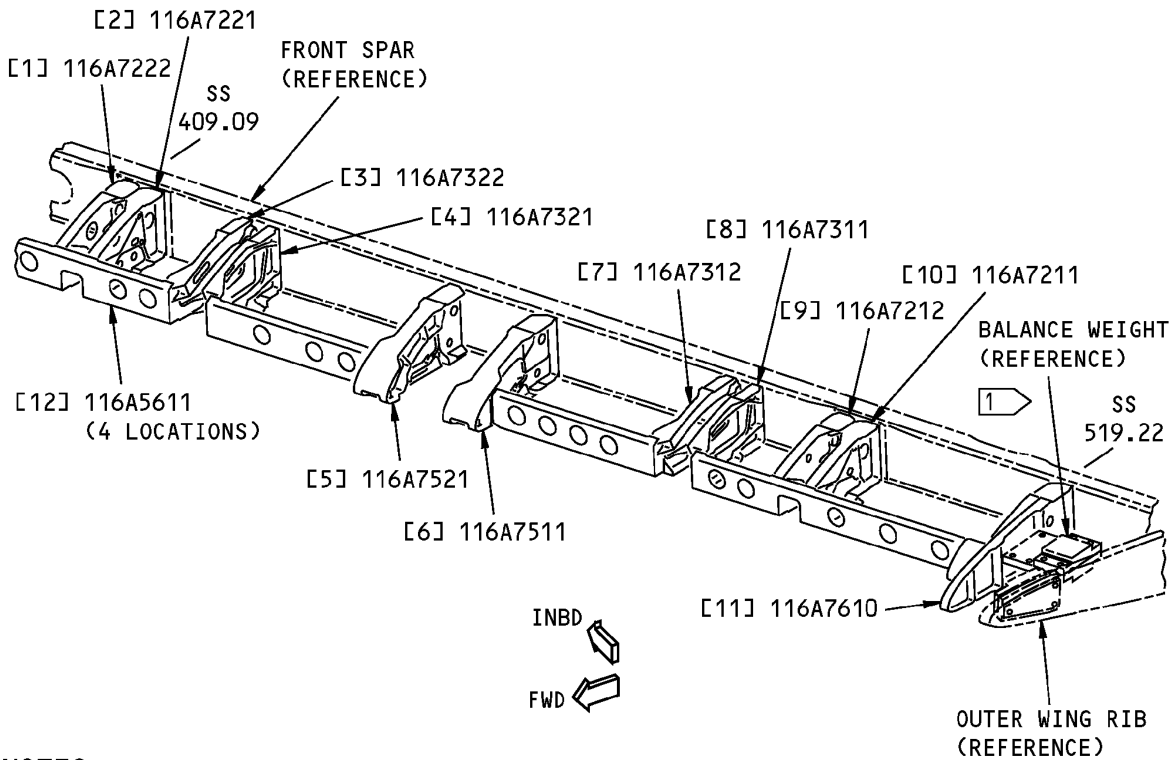
**Wing Outboard Fixed Leading Edge Slat Number 3 Structure Identification**
**Figure 4**




<u>NOTE</u>: REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

SLAT NUMBER 2 IS SHOWN, SLAT NUMBER 7 IS OPPOSITE

**Wing Outboard Fixed Leading Edge Slat Number 2 Structure Identification**
**Figure 5**




NOTES

- REFER TO TABLE 6 FOR THE LIST OF MATERIALS.

1 FOR CUM LINE NUMBER 789 AND ON WITH WINGLET FULL PROVISION INSTALLATION.

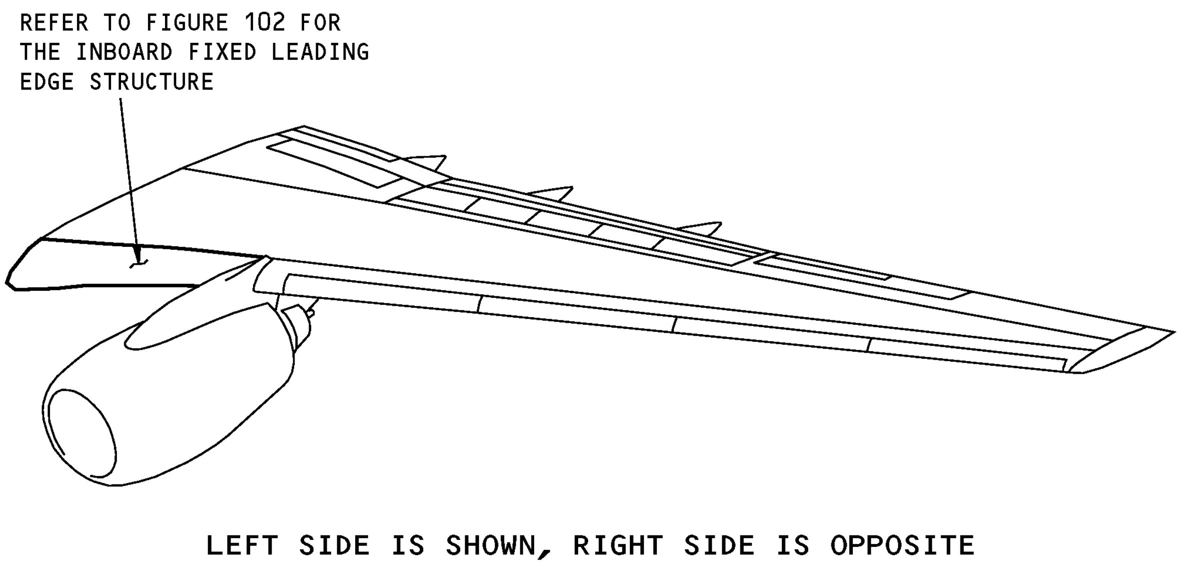
SLAT NUMBER 1 IS SHOWN, SLAT NUMBER 8 IS OPPOSITE

**Wing Outboard Fixed Leading Edge Slat Number 1 Structure Identification**
**Figure 6**

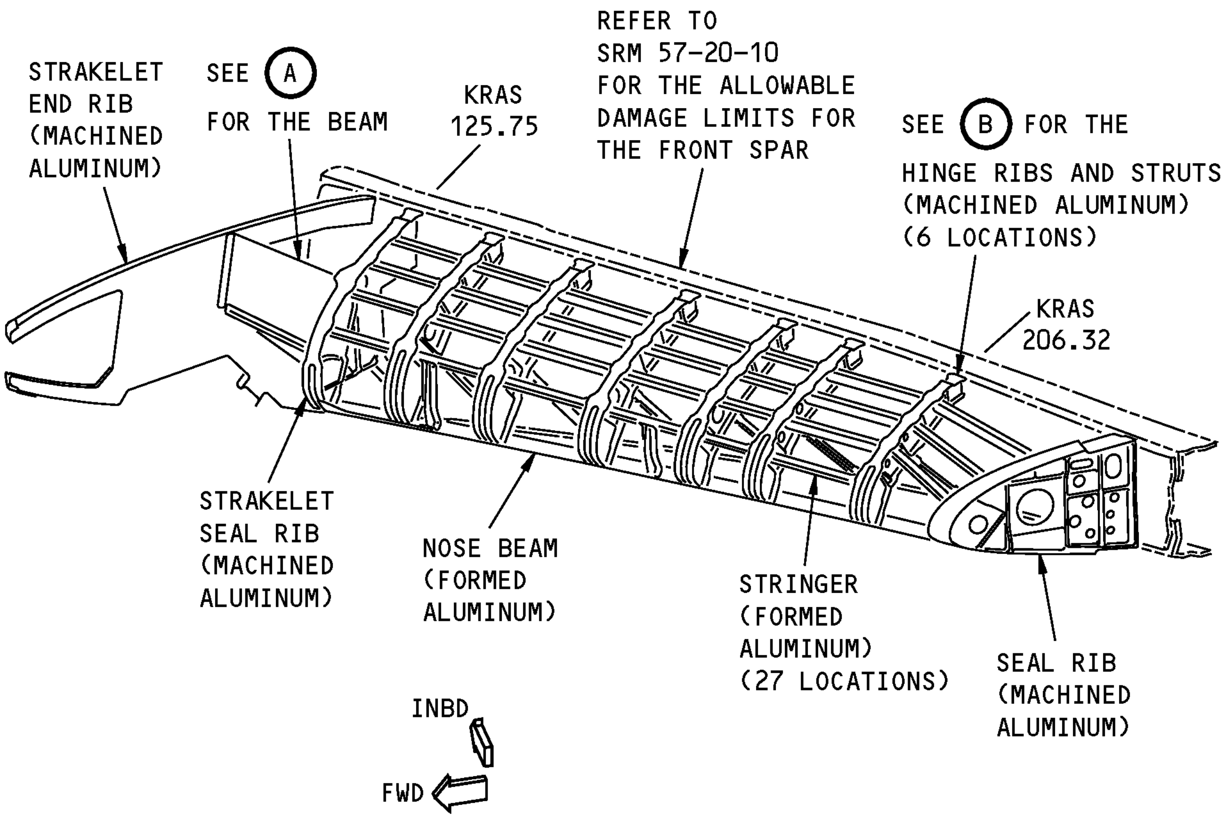
## ALLOWABLE DAMAGE 1 - WING INBOARD FIXED LEADING EDGE STRUCTURE

### 1. Applicability

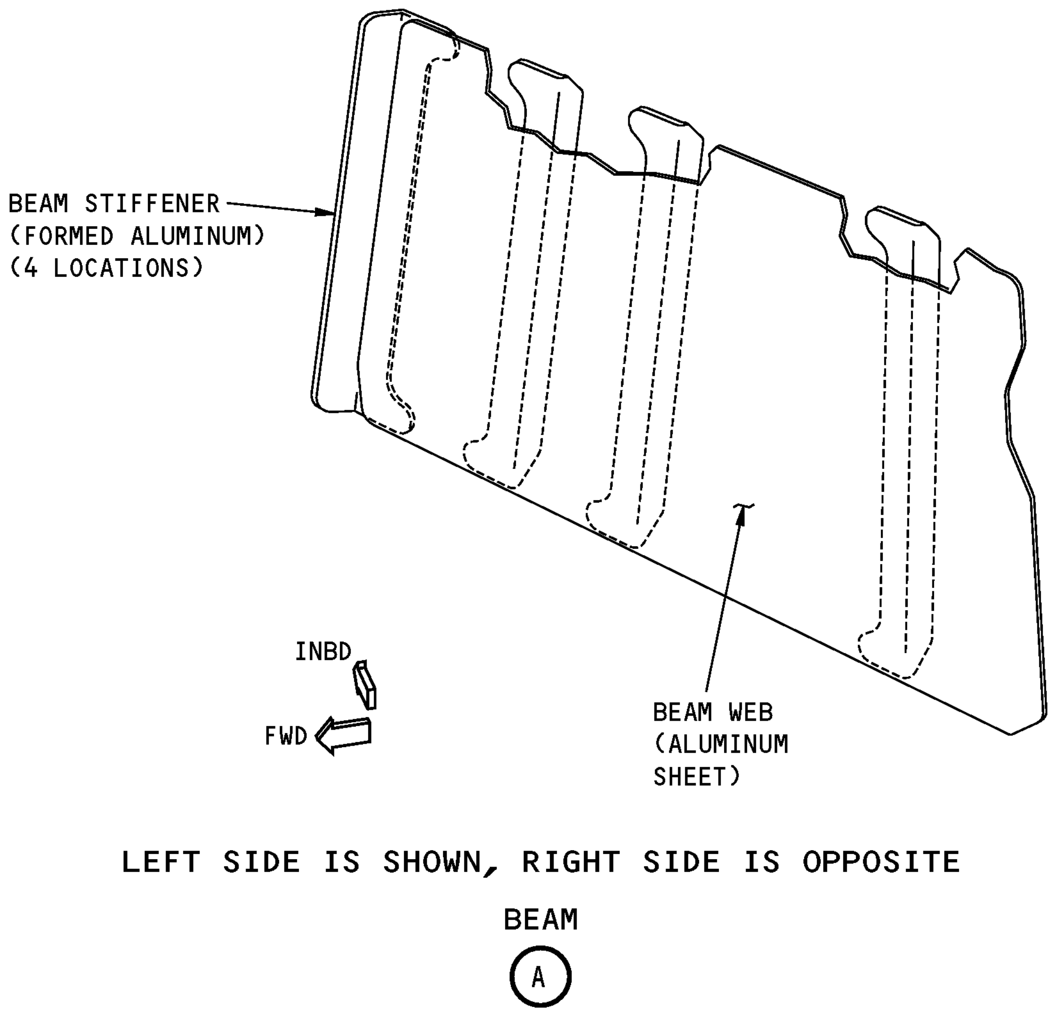
A. This subject gives the allowable damage limits for the wing inboard fixed leading edge structure shown in Wing Inboard Fixed Leading Edge Structure Location, Figure 101/ALLOWABLE DAMAGE 1.



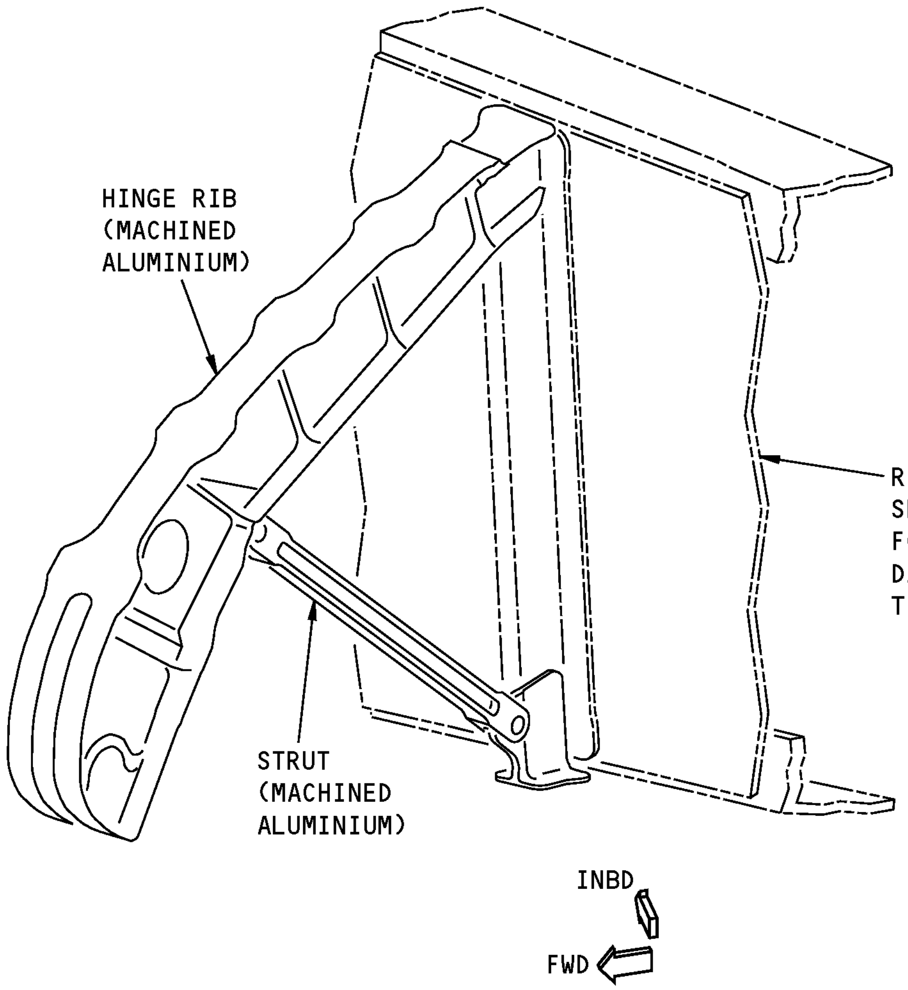

**Wing Inboard Fixed Leading Edge Structure Location**
**Figure 101**




**Wing Inboard Fixed Leading Edge Structure**
**Figure 102 (Sheet 1 of 3)**




**Wing Inboard Fixed Leading Edge Structure**
**Figure 102 (Sheet 2 of 3)**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

TYPICAL HINGE RIB

B

**Wing Inboard Fixed Leading Edge Structure**
**Figure 102 (Sheet 3 of 3)**

## 2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

(1) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(2) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Ribs and Struts

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , D , E , F , H , and I .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Stringers

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and H.

(3) Dents are not permitted.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter

(b) A minimum of 4.0 inches away from other damage

(c) A minimum of 1.0 inch away from a fastener hole

(d) A minimum of 0.50 inch away from a material edge

(e) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

1) Install the rivet without sealant.

C. Beam

(1) Webs

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and J .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , E , F , H , and J .

(c) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail G .

(d) Holes and Punctures are permitted if they are:

1) A maximum of 0.25 inch in diameter

2) A minimum of 4.0 inches away from other damage

3) A minimum of 1.0 inch away from a fastener hole

4) A minimum of 0.50 inch away from a material edge

5) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

a) Install the rivet without sealant.

(2) Stiffeners

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , E , F , and I .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

D. Nose Beam

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and J .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , E , F , H , and J .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail G .

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter

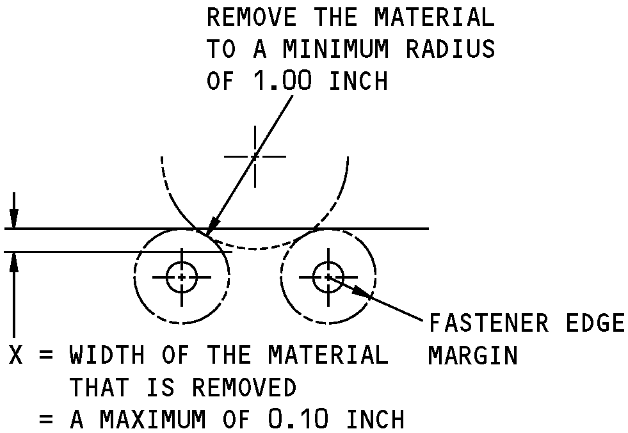
(b) A minimum of 4.0 inches away from other damage

(c) A minimum of 1.0 inch away from a fastener hole

(d) A minimum of 0.50 inch away from a material edge

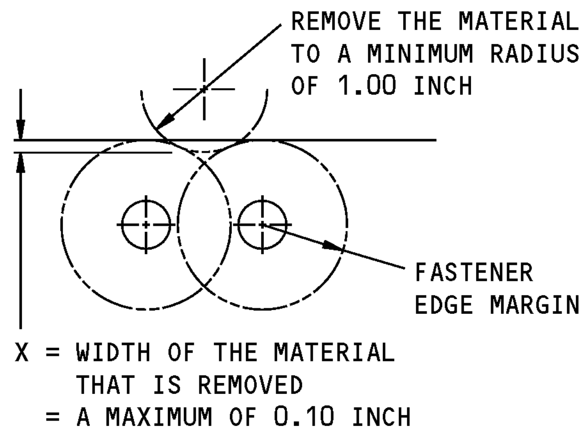
(e) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

1) Install the rivet without sealant.



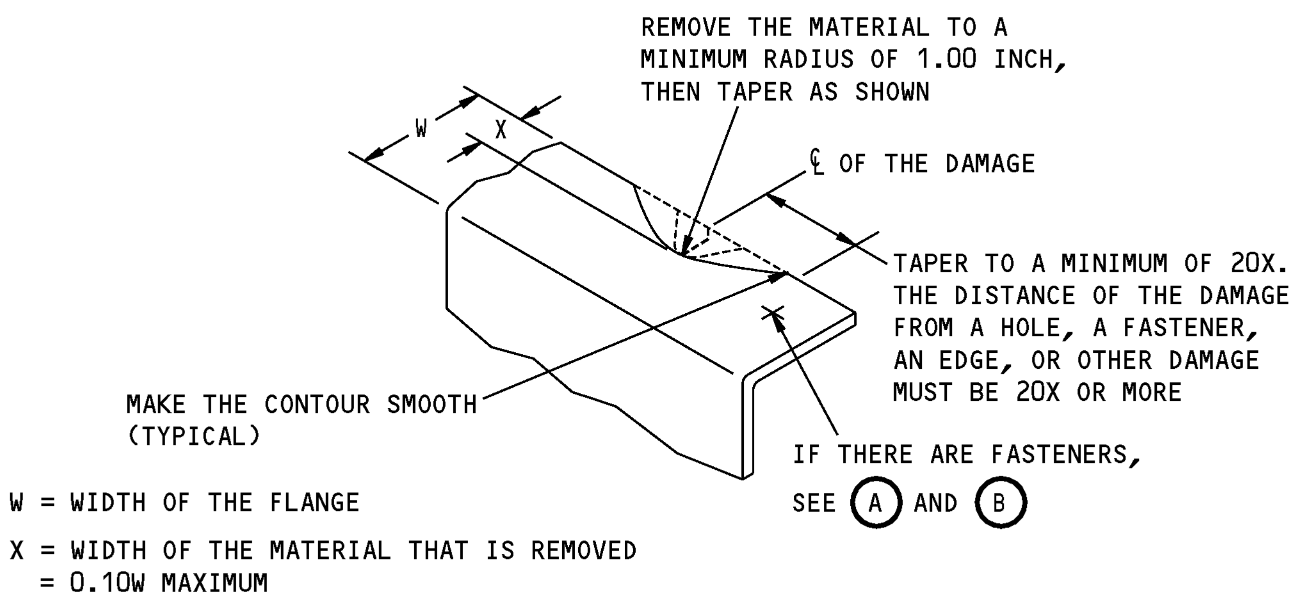

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

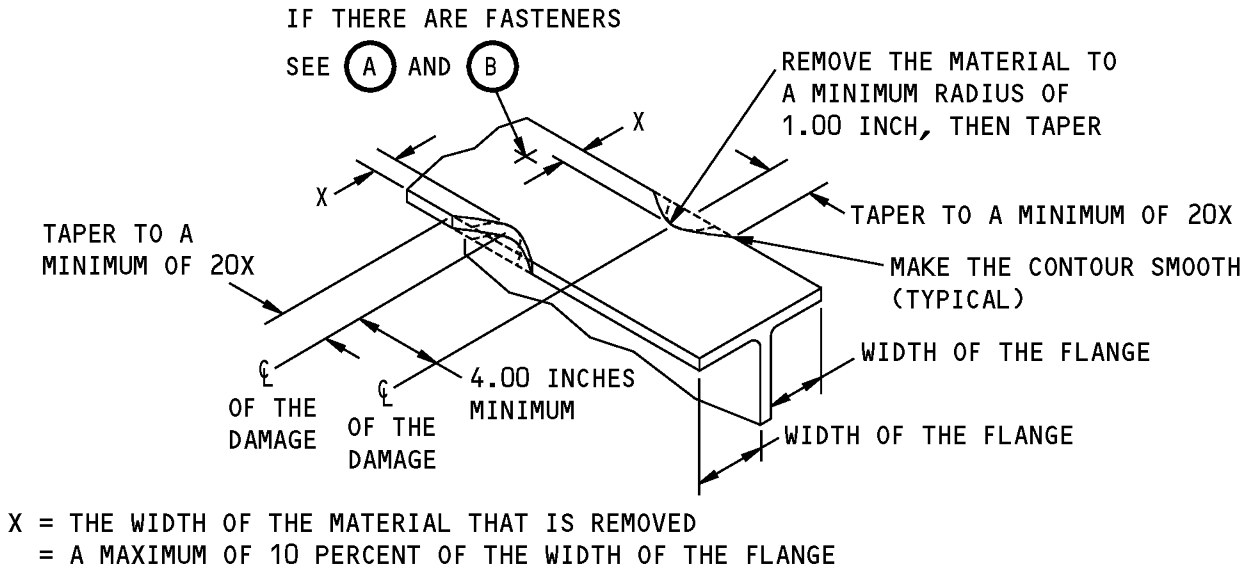
B




REMOVAL OF DAMAGED MATERIAL ON AN EDGE

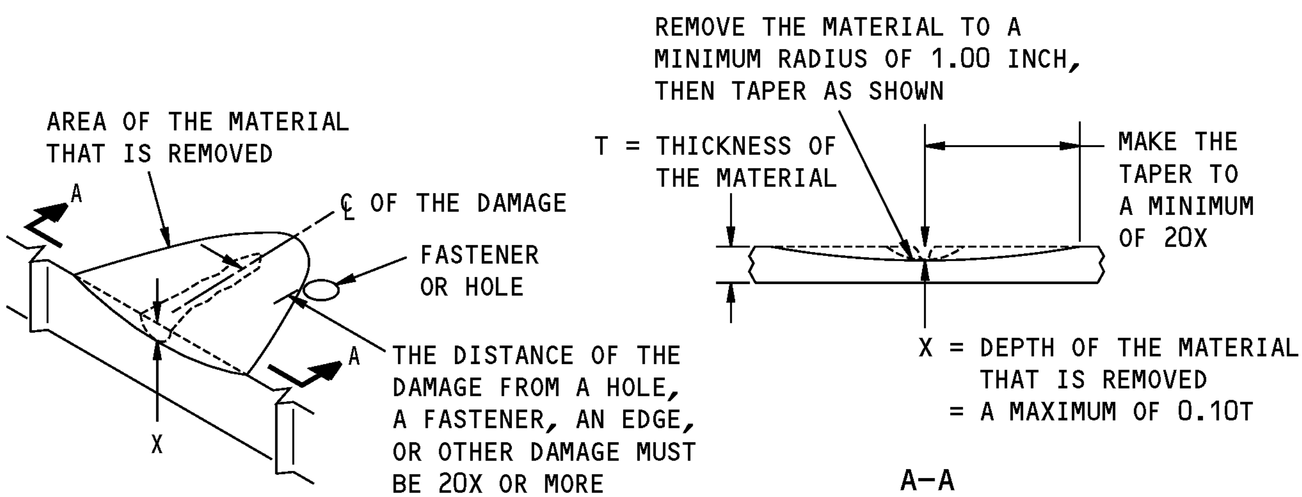
C

**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 8)**




REMOVAL OF DAMAGED MATERIAL ON AN EDGE

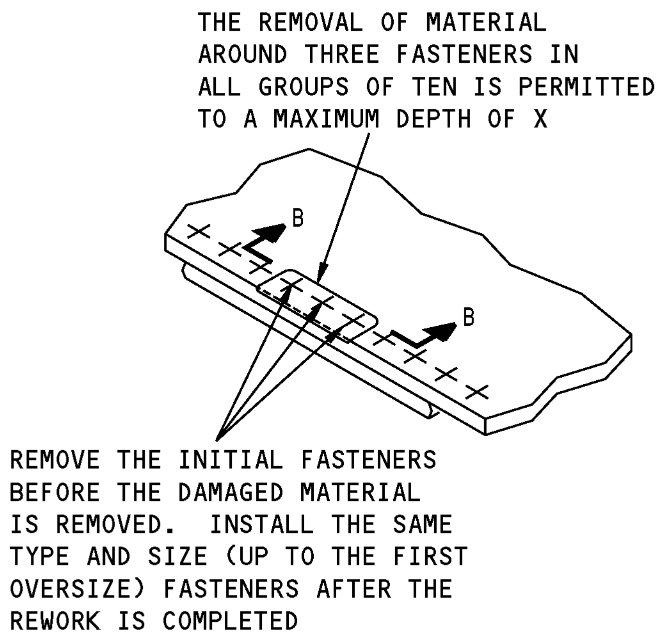
D




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

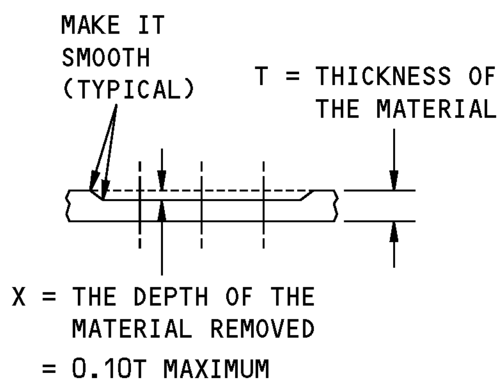
E

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 8)**



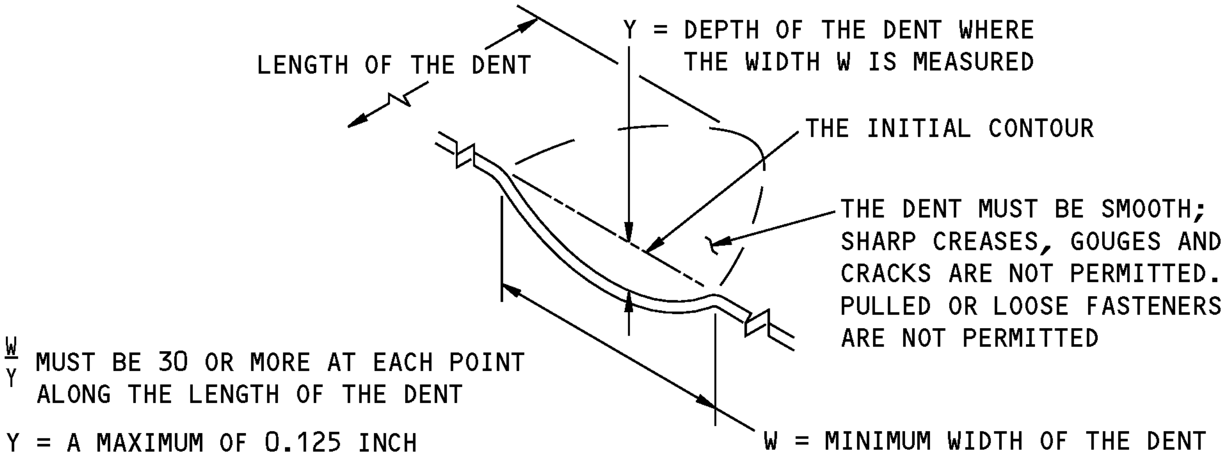

REMOVAL OF CORROSION AROUND THE FASTENERS

F



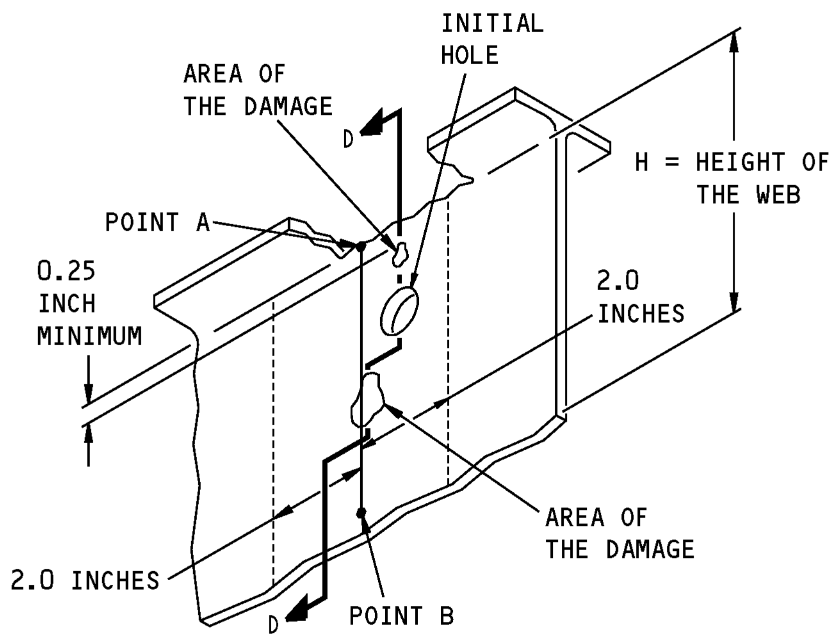

B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 8)**




DENT THAT IS PERMITTED

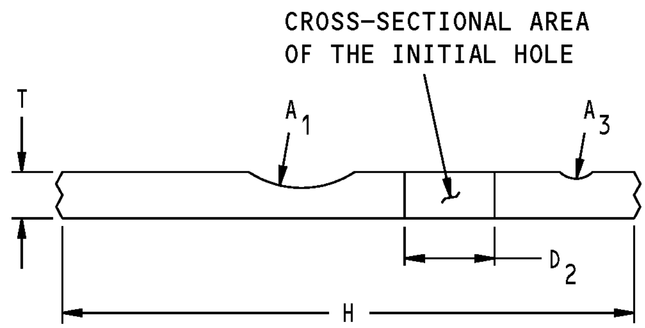
G




REMOVAL OF DAMAGED MATERIAL FROM A WEB
(A TEE SECTION IS SHOWN)

H

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 8)**




$D_2$ = DIAMETER OF THE HOLE AT LOCATION 2

H = HEIGHT OF THE WEB

T = THICKNESS OF THE WEB

$A_i$ = INITIAL AREA OF THE WEB
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $HT - D_2T$

$A_1$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

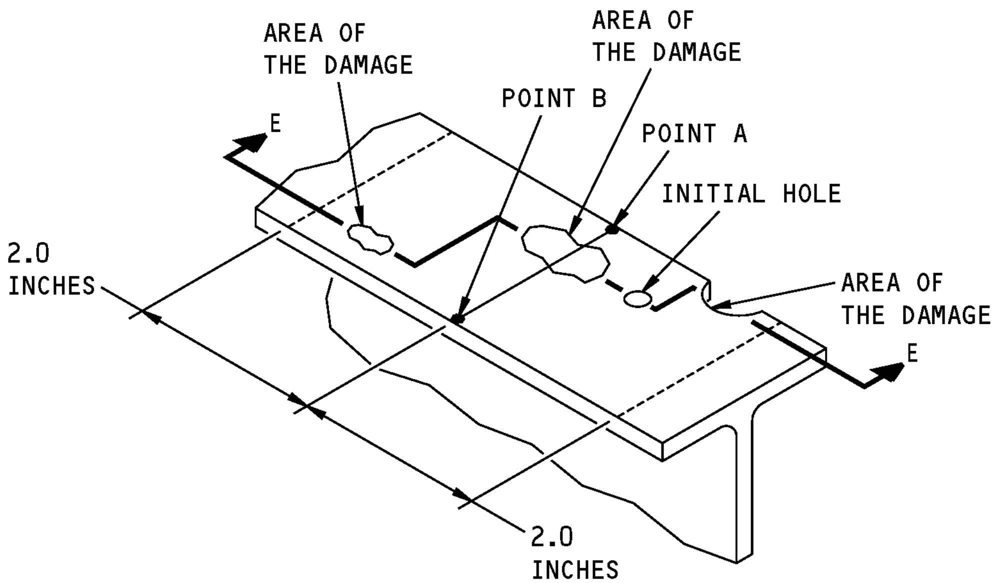
$A_3$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 3

$$\left(\frac{A_1 + A_3}{A_i}\right) \times 100 = \text{PERCENT OF CROSS-SECTIONAL AREA REMOVED}$$
$$= \text{A MAXIMUM OF 10 PERCENT}$$

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE WEB.

(ROTATED 90° CLOCKWISE)
D-D

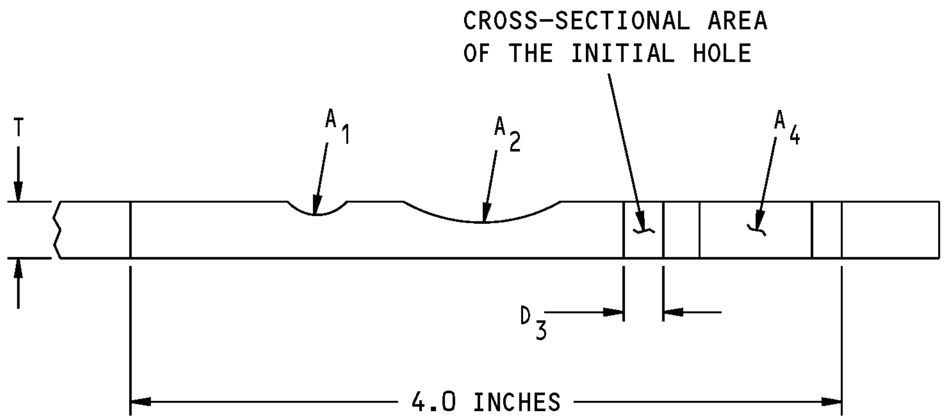
**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 8)**




REMOVAL OF DAMAGE MATERIAL ON A FLANGED SURFACE
(A TEE SECTION IS SHOWN)

I

**Allowable Damage Limits**
**Figure 103 (Sheet 6 of 8)**




$D_3$ = DIAMETER OF THE HOLE AT LOCATION 3

T = THICKNESS OF THE FLANGE

$A_i$ = INITIAL AREA OF THE FLANGE
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $4T - D_3T$

$A_1$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

$A_2$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 2

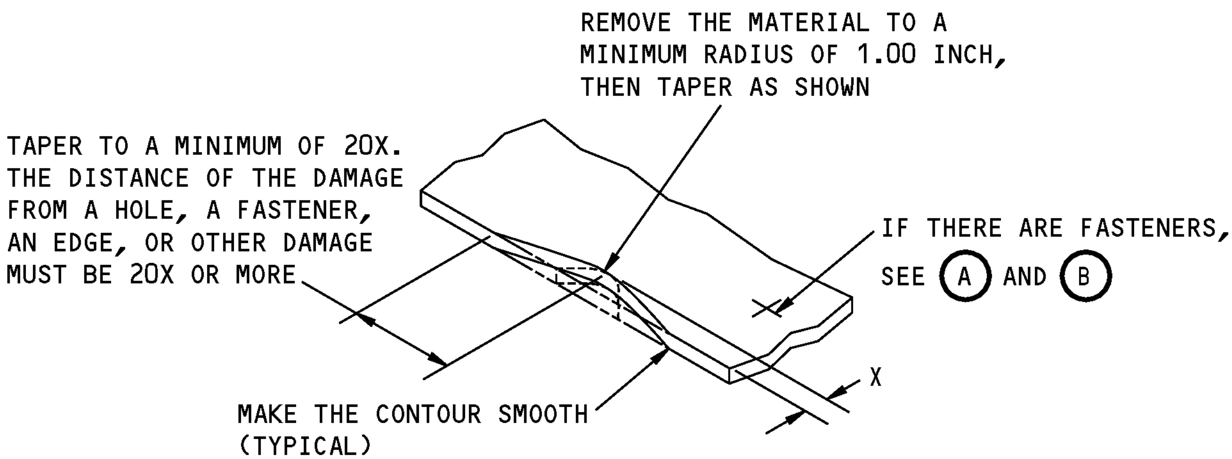
$A_4$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 4

$$\left(\frac{A_1 + A_2 + A_4}{A_i}\right) \times 100$$ = PERCENT OF CROSS-SECTIONAL AREA REMOVED
= A MAXIMUM OF 10 PERCENT

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE FLANGE SURFACE.

E-E

**Allowable Damage Limits**
**Figure 103 (Sheet 7 of 8)**




X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL WEB

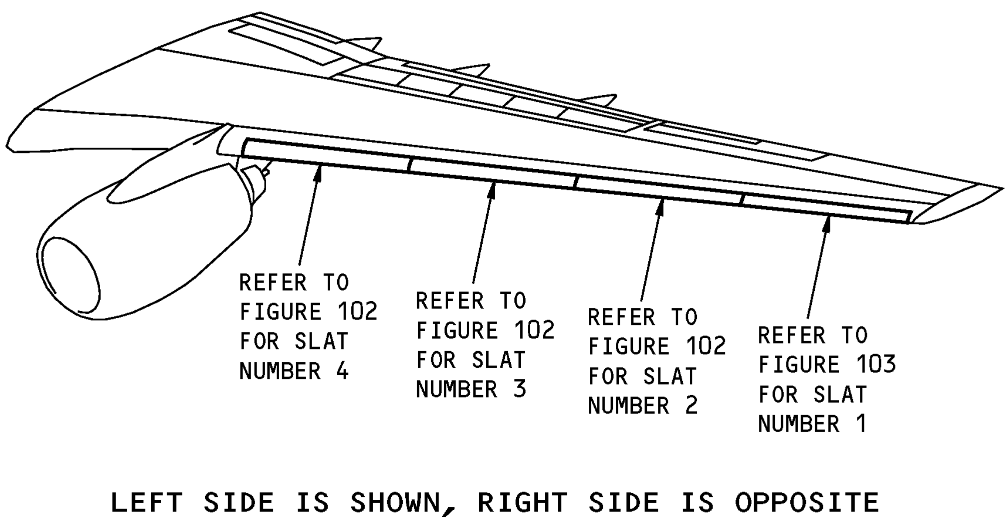
J

**Allowable Damage Limits**
**Figure 103 (Sheet 8 of 8)**

## ALLOWABLE DAMAGE 2 - WING OUTBOARD FIXED LEADING EDGE STRUCTURE

### 1. Applicability

A. This subject gives the allowable damage limits for the wing outboard fixed leading edge structure shown in Wing Outboard Fixed Leading Edge Structure Locations, Figure 101/ALLOWABLE DAMAGE 2.




**Wing Outboard Fixed Leading Edge Structure Locations**
**Figure 101**

### 2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

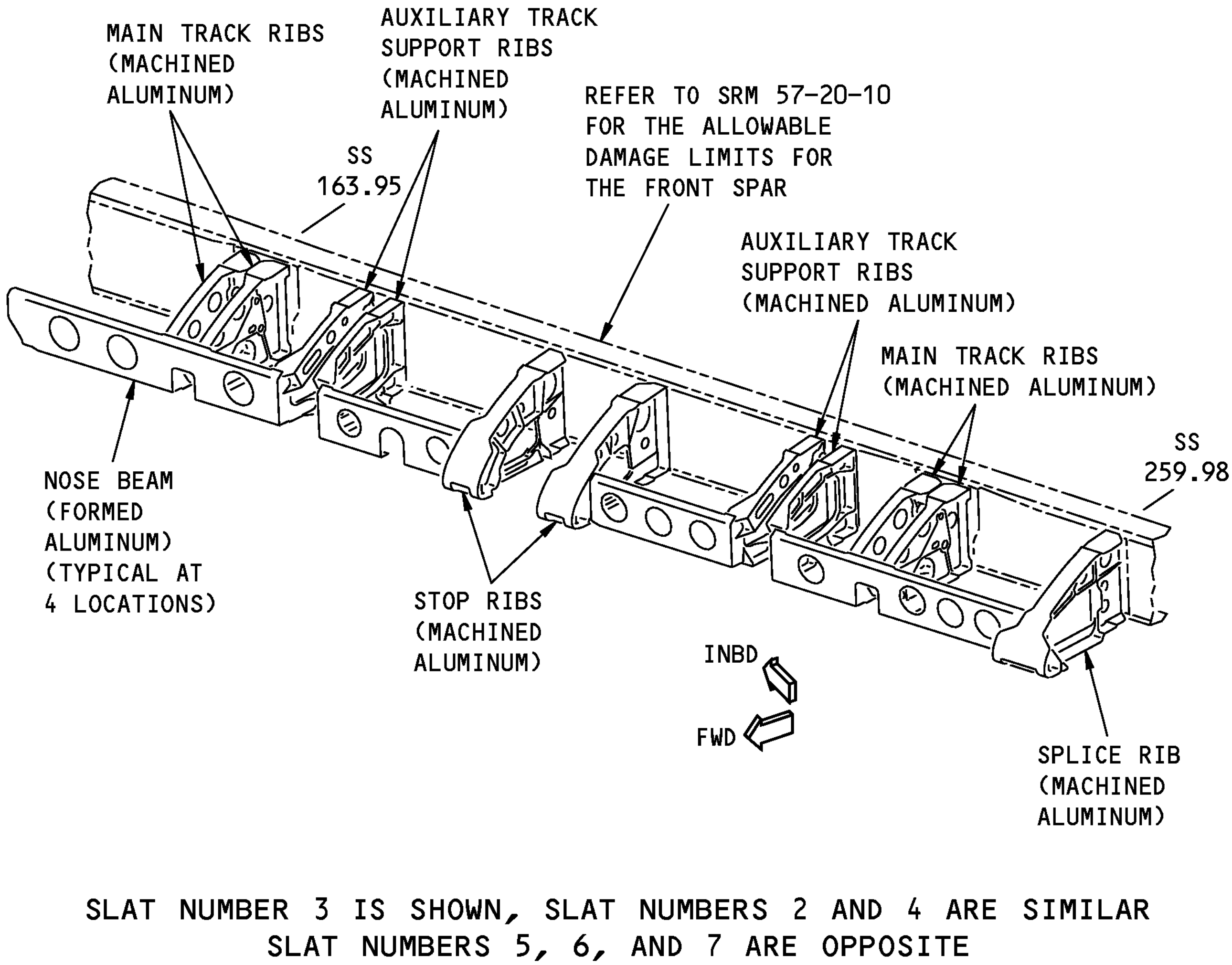
(1) Flap peen or shot peen the reworked areas of the main track ribs.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

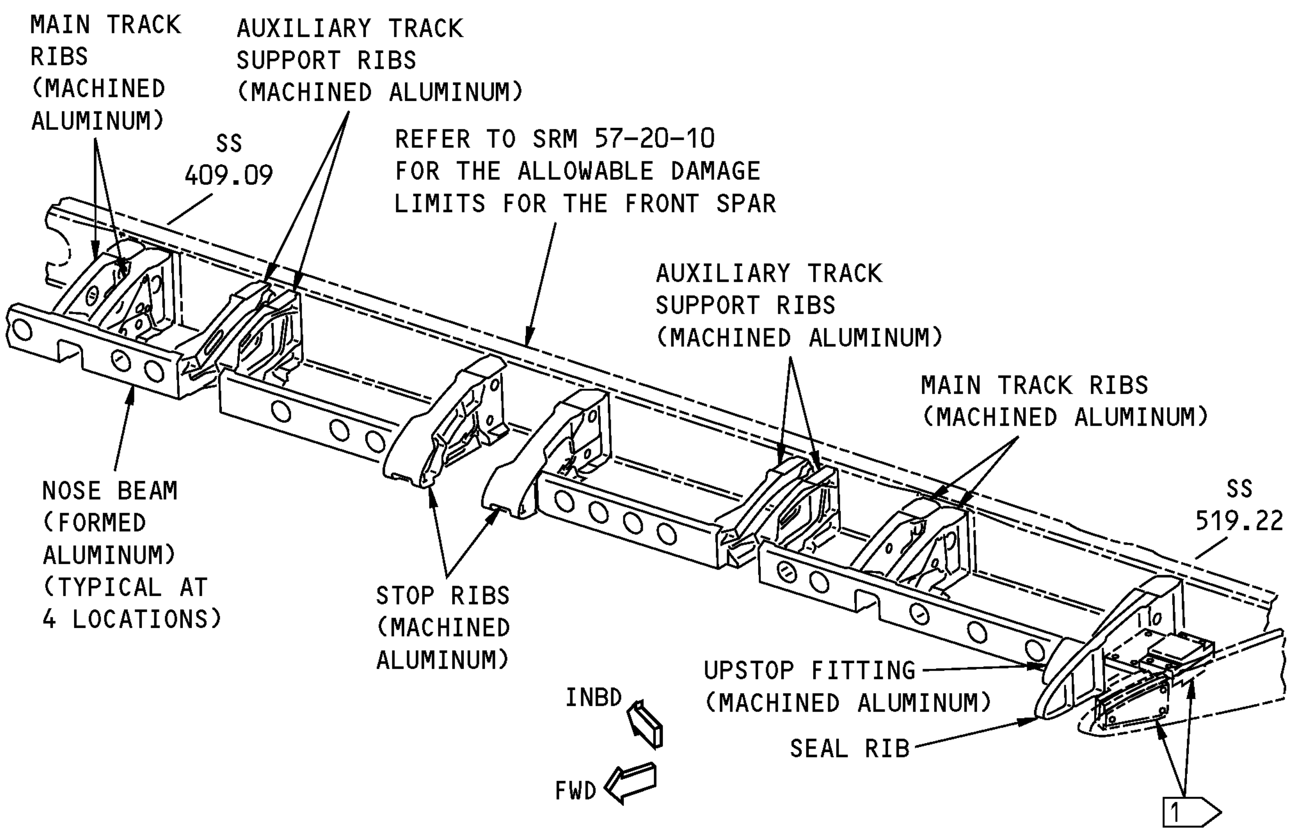
(2) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(3) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.




G29036 S0006594610_V2

**Wing Outboard Fixed Leading Edge Slat Number 3 Structure**
**Figure 102**




NOTES

1 FOR CUM LINE NUMBER 789 AND ON WITH FULL WINGLET PROVISION INSTALLATION.

SLAT NUMBER 1 IS SHOWN, SLAT NUMBER 8 IS OPPOSITE

**Wing Outboard Fixed Leading Edge Slat Number 1 Structure**
**Figure 103**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Ribs and Upstop Fitting

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , D , and H .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , D , E , F , H , I , and J .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Nose Beam

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , C , E , and F .

(3) Dents are permitted as given in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail G .

(4) Holes and Punctures:

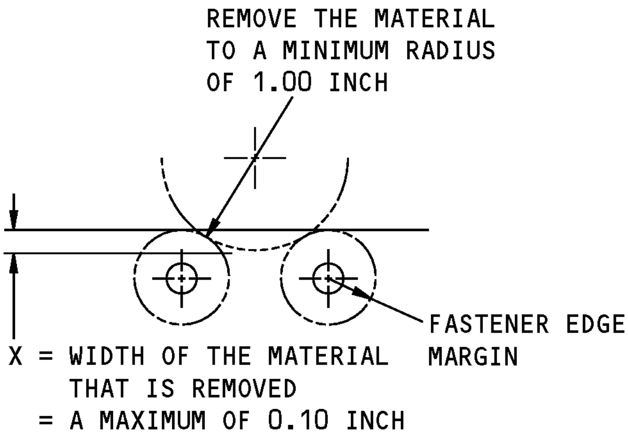
(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter.

2) A minimum of 4.0 inches away from other damage.

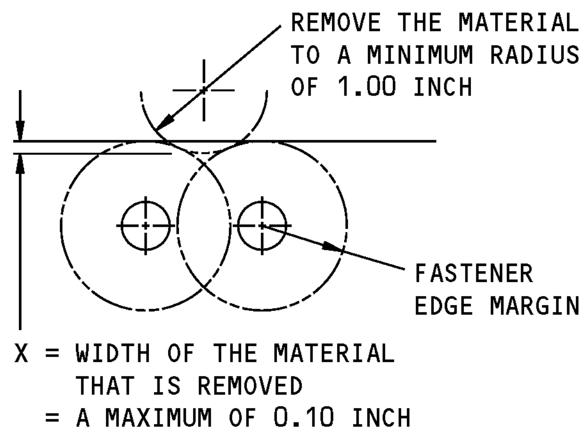
3) A minimum of 1.0 inch away from a fastener hole.

4) A minimum of 0.50 inch away from a material edge.
5) Filled with a 2117-T3 or 2117-T4 protruding head rivet.
   a) Install the rivet without sealant.



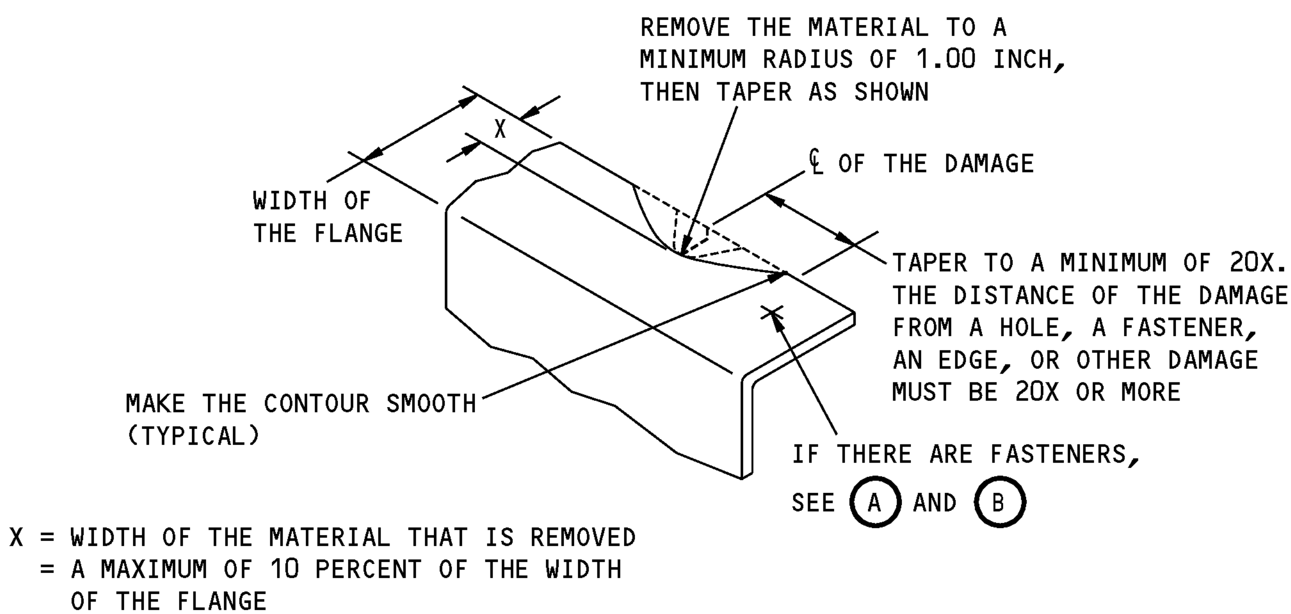

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

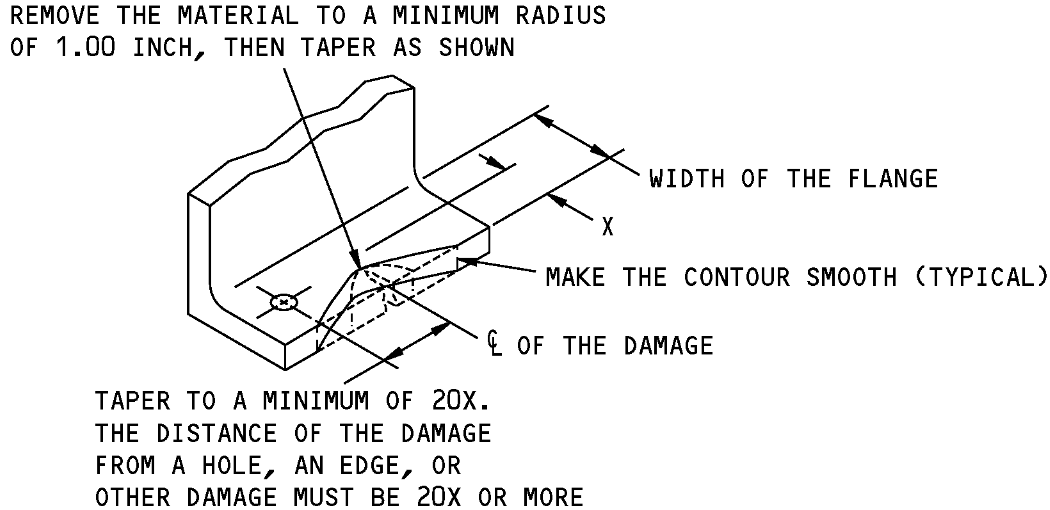
B




REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

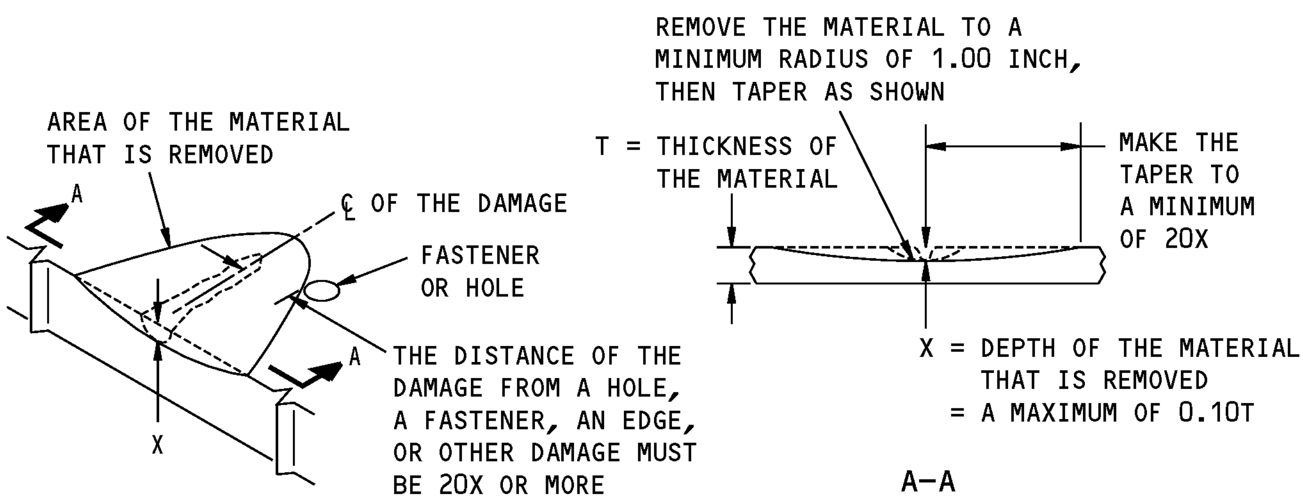
**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 7)**




X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

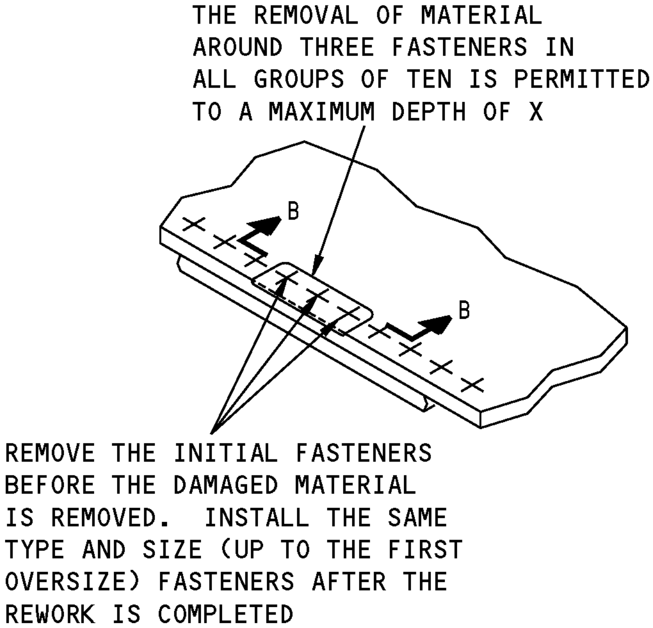
D




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

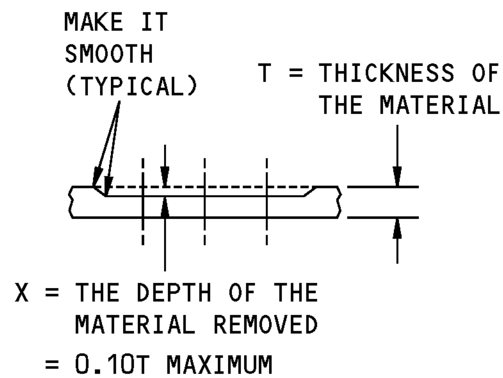
E

**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 7)**

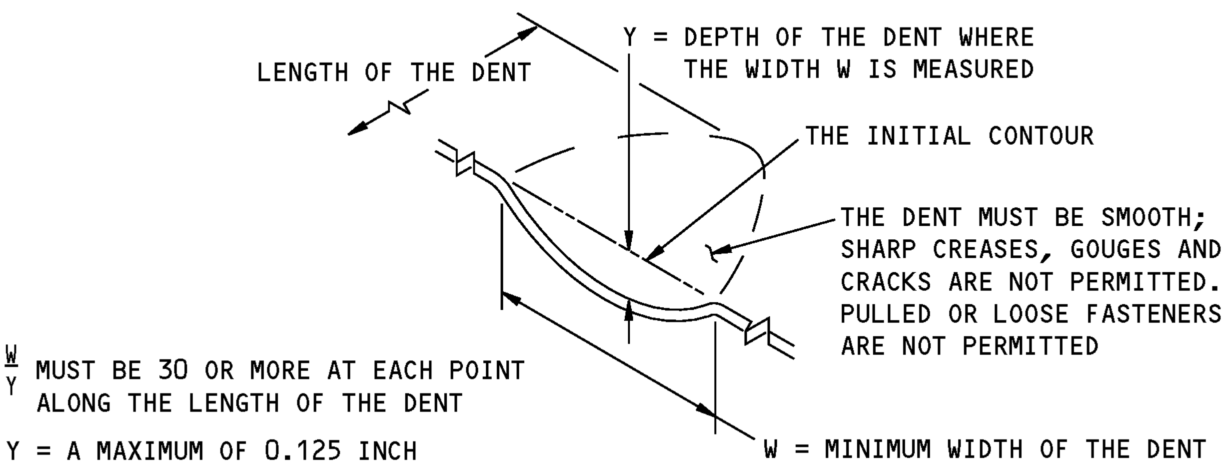



REMOVAL OF CORROSION AROUND THE FASTENERS

F



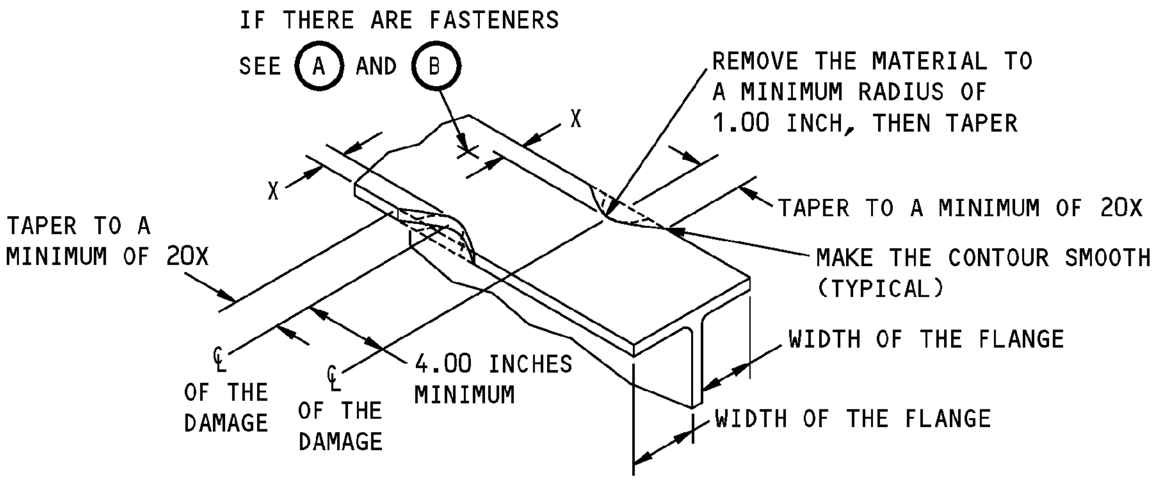

B-B




DENT THAT IS PERMITTED

G

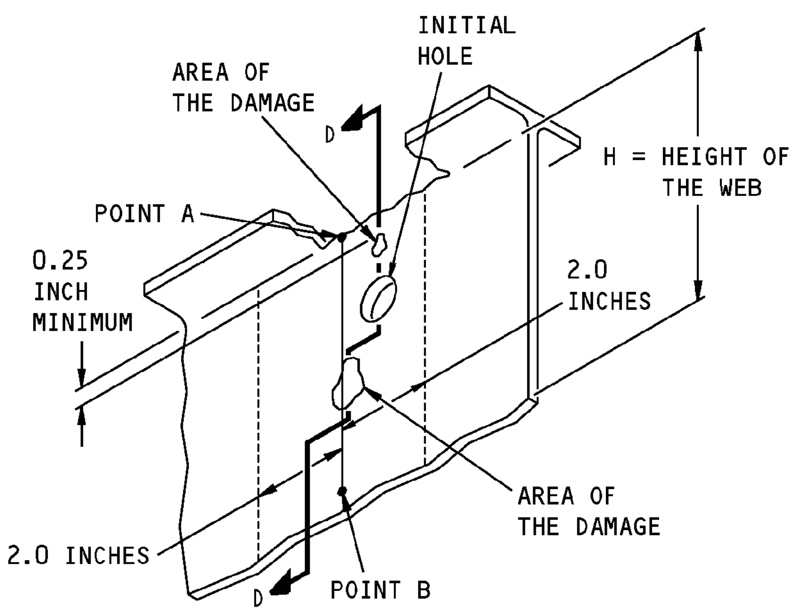
**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 7)**




X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

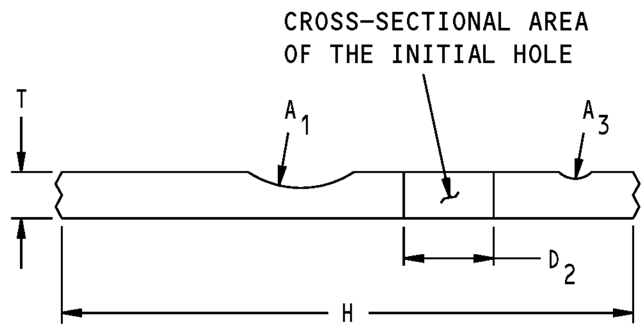
H




REMOVAL OF DAMAGED MATERIAL FROM A WEB OF A TEE SECTION

I

**Allowable Damage Limits**
**Figure 104 (Sheet 4 of 7)**




$D_2$ = DIAMETER OF THE HOLE AT LOCATION 2

H = HEIGHT OF THE WEB

T = THICKNESS OF THE WEB

$A_i$ = INITIAL AREA OF THE WEB
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $HT - D_2T$

$A_1$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

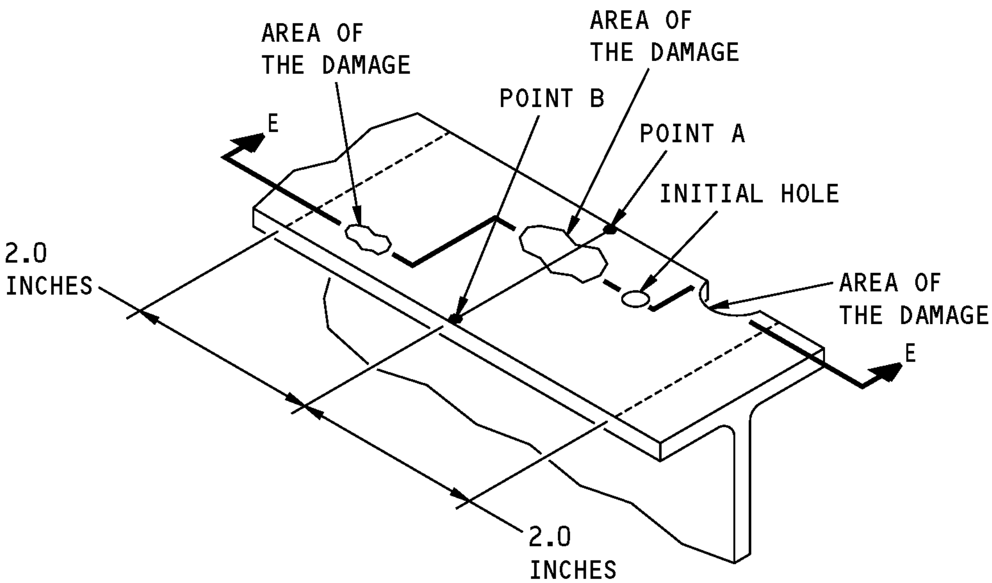
$A_3$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 3

$\left(\frac{A_1 + A_3}{A_i}\right)$ X 100 = PERCENT OF CROSS-SECTIONAL AREA REMOVED
= A MAXIMUM OF 10 PERCENT

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE WEB.

(ROTATED 90° CLOCKWISE)
D-D

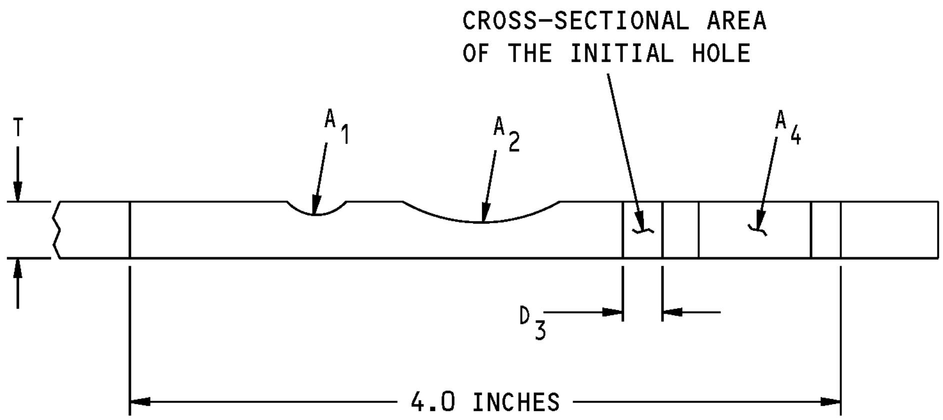
**Allowable Damage Limits**
**Figure 104 (Sheet 5 of 7)**




REMOVAL OF DAMAGE MATERIAL ON A FLANGED SURFACE
(A TEE SECTION IS SHOWN)

J

**Allowable Damage Limits**
**Figure 104 (Sheet 6 of 7)**




$D_3$ = DIAMETER OF THE HOLE AT LOCATION 3

T = THICKNESS OF THE FLANGE

$A_i$ = INITIAL AREA OF THE FLANGE
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $4T - D_3T$

$A_1$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

$A_2$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 2

$A_4$ = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 4

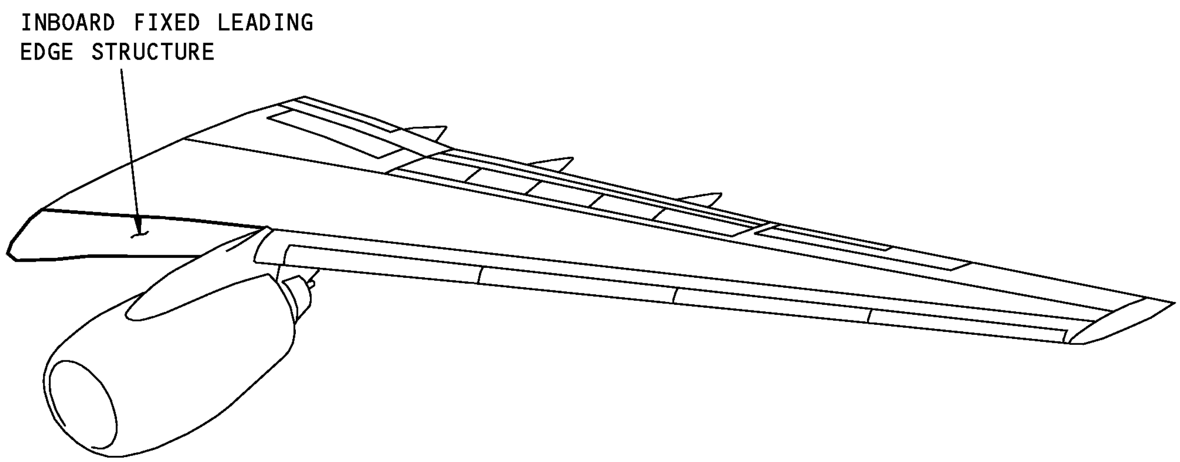
$\left(\frac{A_1 + A_2 + A_4}{A_i}\right)$ X 100 = PERCENT OF CROSS-SECTIONAL AREA REMOVED
= A MAXIMUM OF 10 PERCENT

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE FLANGE SURFACE.

E-E

**Allowable Damage Limits**
**Figure 104 (Sheet 7 of 7)**

## REPAIR 1 - WING INBOARD FIXED LEADING EDGE STRUCTURE

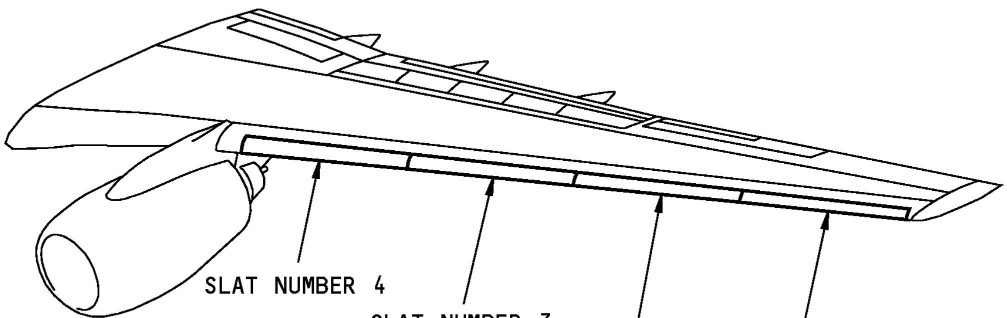



NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Inboard Fixed Leading Edge Structure Location**
**Figure 201**

## REPAIR 2 - WING OUTBOARD FIXED LEADING EDGE STRUCTURE

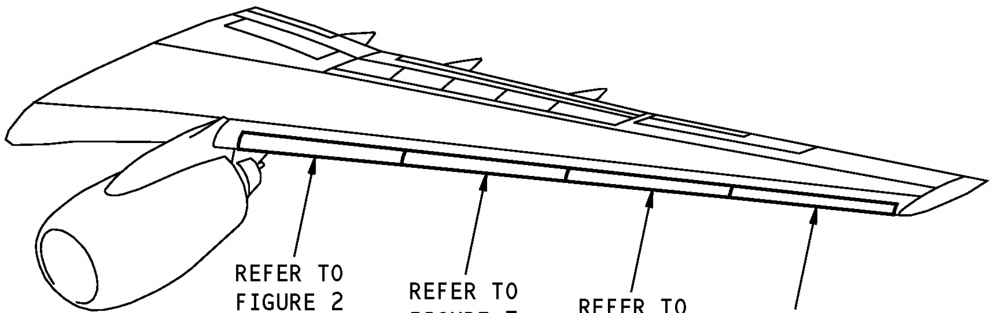



NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Outboard Fixed Leading Edge Structure Locations**
**Figure 201**

## IDENTIFICATION 1 - WING LEADING EDGE SLAT SKIN



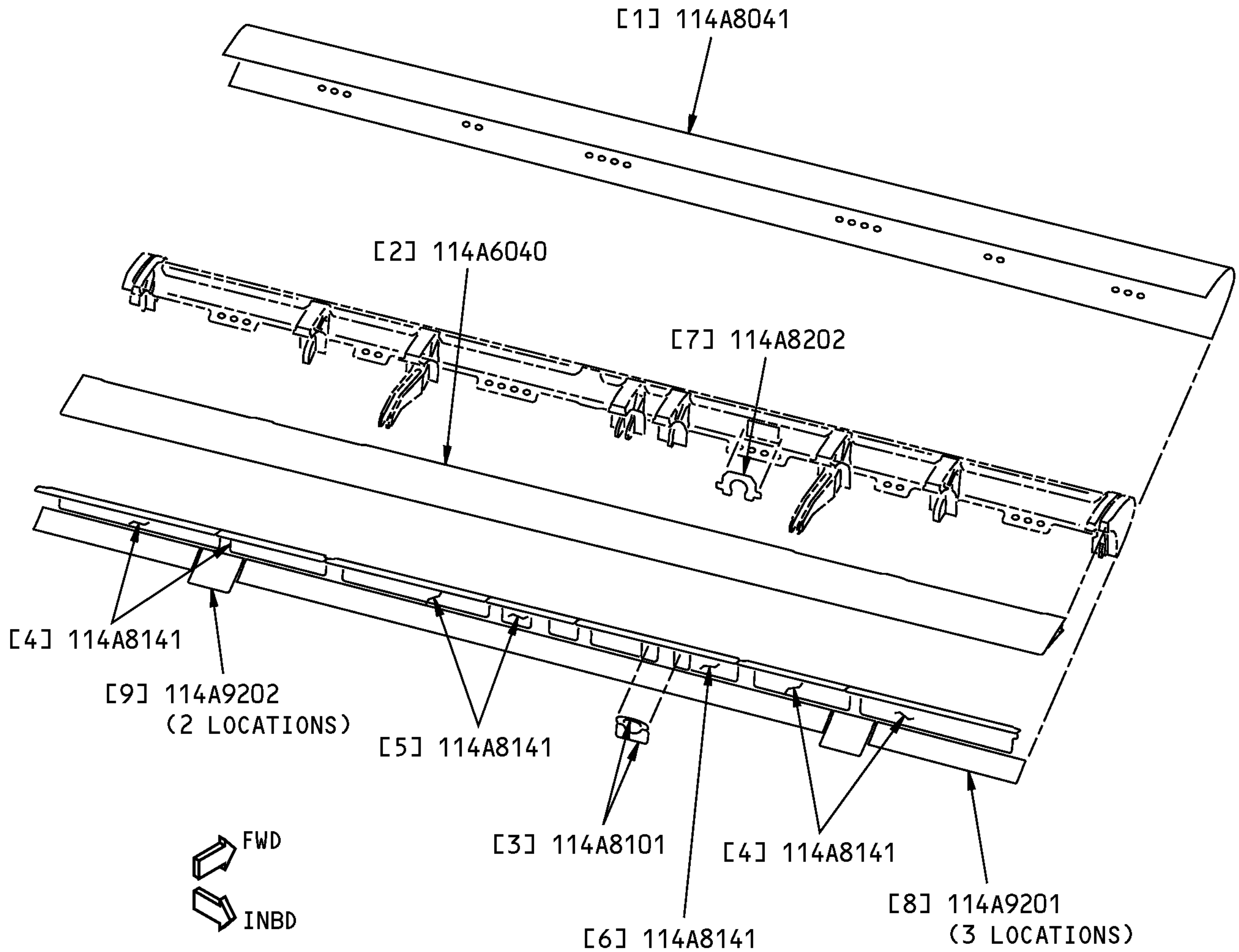

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Leading Edge Slat Skin Locations**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 114A0011 | Left Wing, Slat Functional Product Collector |
| 114A0012 | Right Wing, Slat Functional Product Collector |
| 114A4002 | Slat Installation, Outboard Leading Edge, Wing |
| 114A5010 | Slat Assembly - No. 1 and 8 |
| 114A5020 | Slat Assembly - No. 2 and 7 |
| 114A5030 | Slat Assembly - No. 3 and 6 |
| 114A5040 | Slat Assembly - No. 4 and 5 |



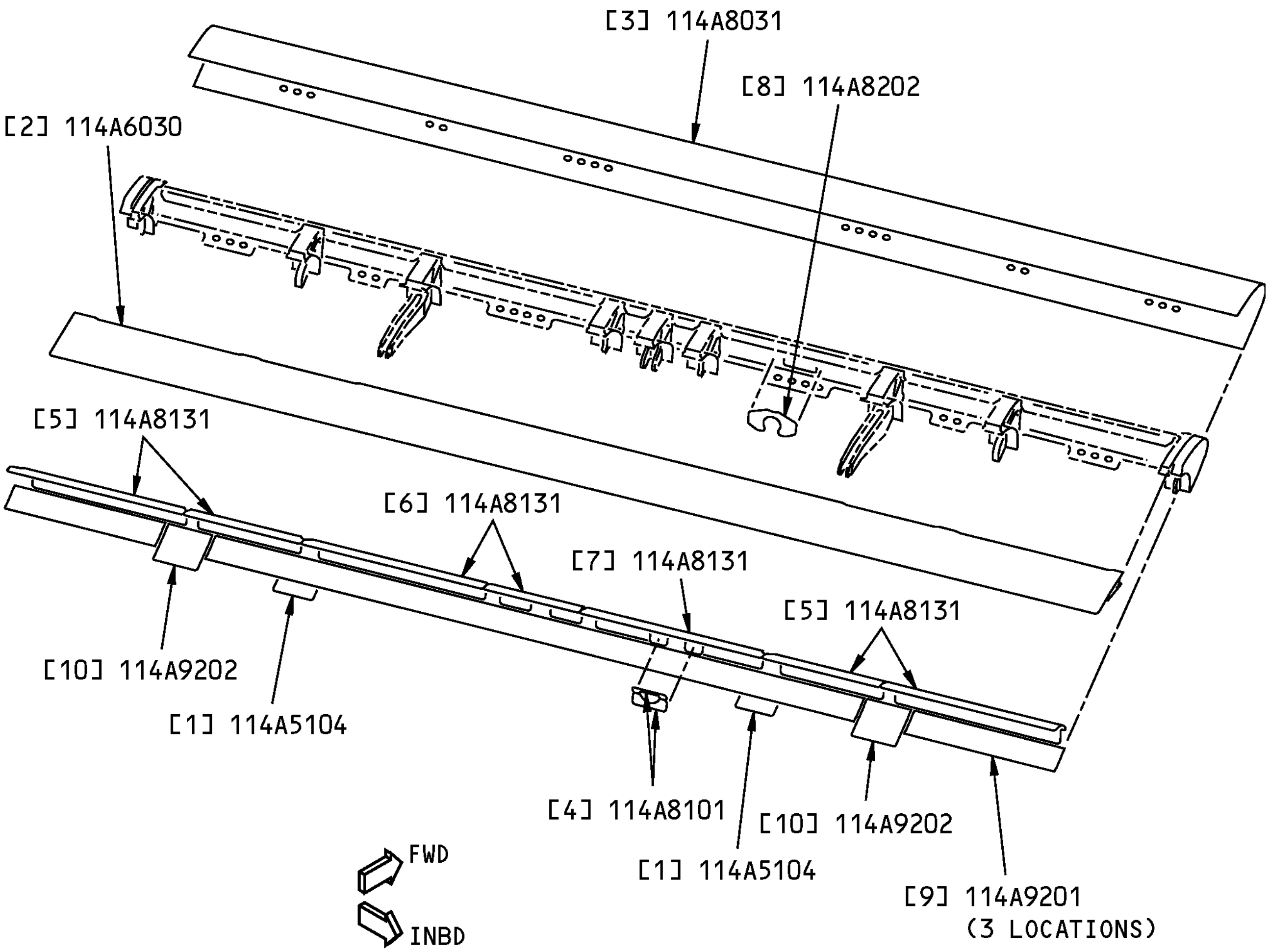

NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Wing Leading Edge Slat Number 4 and Number 5 Skin Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Nose Skin | 0.063 (1.6) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [2] | Wedge Assembly | | | LINE NUMBER 1 TO 19 |
| | Upper Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Upper Doubler | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core (2) | | Aluminum honeycomb as given in BMS 4-4, Type 3-15NPA, Grade I. Refer to Figure 6 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-10NPA, Grade I. Refer to Figure 6 | |
| [2] | Wedge Assembly | | | LINE NUMBER 20 AND ON |
| | Upper Skin | 0.032 (0.81) | 7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7 | |
| [3] | Backup Plate (2) | 0.032 (0.81) | 6013-T4 sheet as given in AMS 4347 | |
| [4] | Cove Skin (4) | 0.050 (1.27) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [5] | Cove Skin (2) | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [6] | Cove Skin Assembly | | | |
| | Cove Skin | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| | Doubler | 0.160 (4.06) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [7] | Cover Plate | 0.050 (1.27) | 7075-T6 sheet as given in QQ-A-250/12 | |
| [8] | Flex Skin (3) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 8 | |
| [9] | Flex Tab (2) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9 | LINE NUMBER 1 TO 19 |
| [9] | Flex Tab (2) | | Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 10 | LINE NUMBER 20 AND ON |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



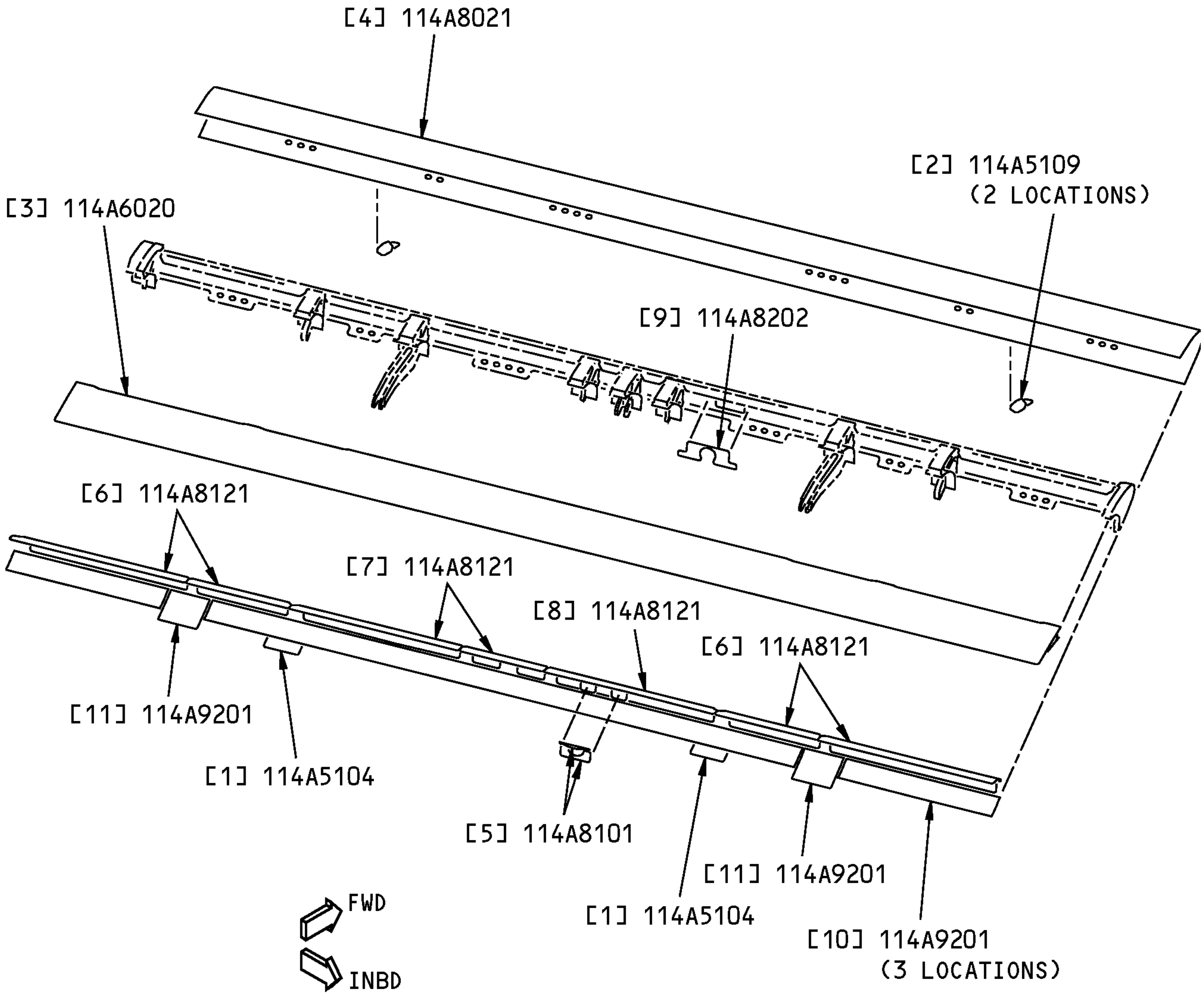

NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

**Wing Leading Edge Slat Number 3 and Number 6 Skin Identification**
**Figure 3**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Skin Tab (2) | 0.100 (2.54) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [2] | Wedge Assembly | | | LINE NUMBER 1 TO 19 |
| | Upper Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Upper Doubler | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core (2) | | Aluminum honeycomb as given in BMS 4-4, Type 3-15NPA, Grade I. Refer to Figure 6 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-10NPA, Grade I. Refer to Figure 6 | |
| [2] | Wedge Assembly | | | LINE NUMBER 20 AND ON |
| | Upper Skin | 0.032 (0.81) | 7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7 | |
| [3] | Nose Skin | 0.063 (1.6) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [4] | Backup Plate (2) | 0.032 (0.81) | 6013-T4 sheet as given in AMS 4347 | |
| [5] | Cove Skin (4) | 0.050 (1.27) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [6] | Cove Skin (2) | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [7] | Cove Skin Assembly | | | |
| | Cove Skin | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| | Doubler | 0.160 (4.06) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [8] | Cover Plate | 0.063 (1.6) | 7075-T6 sheet as given in QQ-A-250/12 | |
| [9] | Flex Skin (3) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 8 | |
| [10] | Flex Tab (2) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9 | LINE NUMBER 1 TO 19 |
| [10] | Flex Tab (2) | | Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 10 | LINE NUMBER 20 AND ON |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



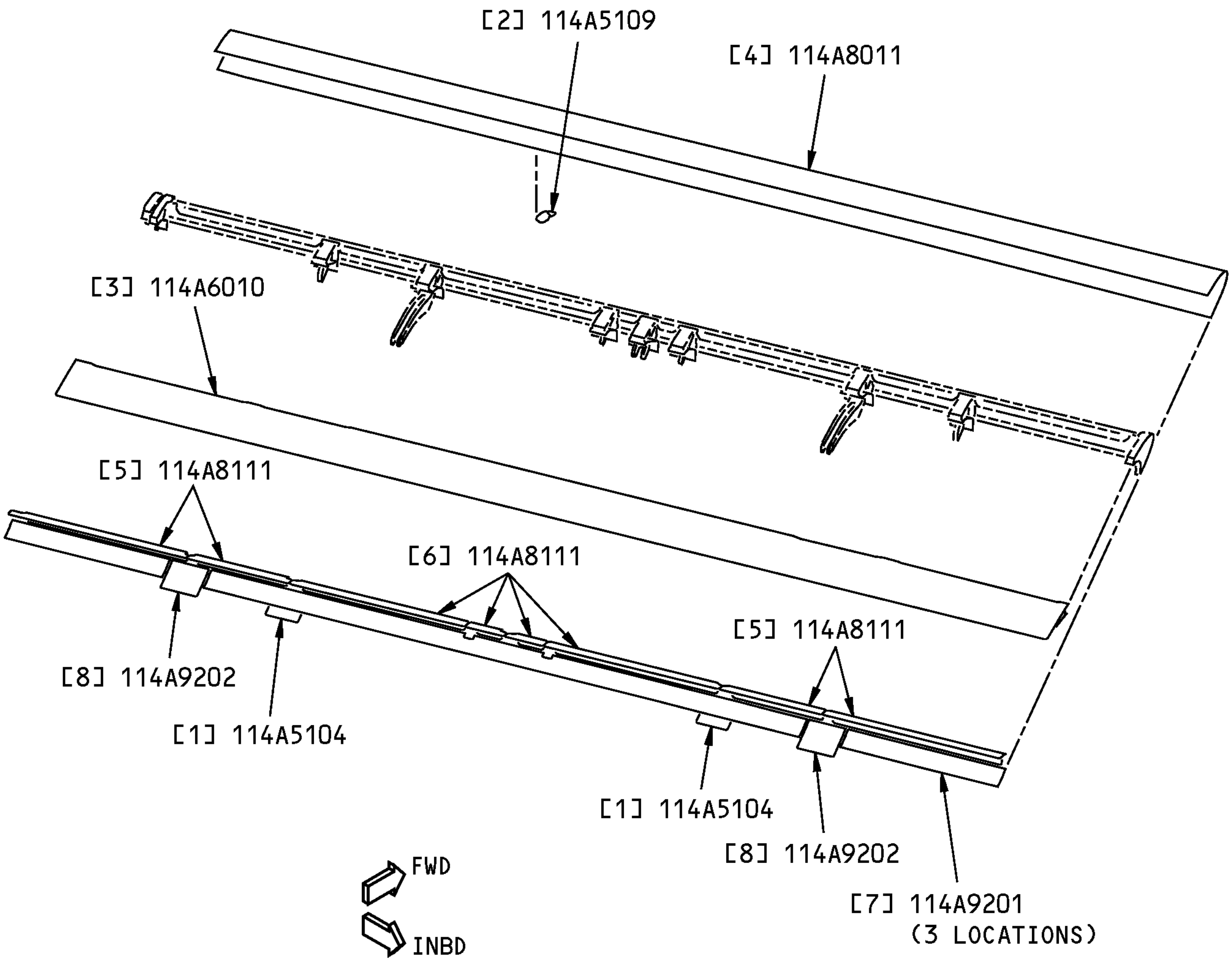

NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

**Wing Leading Edge Slat Number 2 and Number 7 Skin Identification**
**Figure 4**

Table 4:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Skin Tab (2) | 0.100 (2.54) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [2] | Vortilon (2) | | 7050-T7451 plate as given in AMS 4050 (Optional: A356.0-T6 casting as given in AMS 4218) | |
| [3] | Wedge Assembly | | | LINE NUMBER 1 TO 19 |
| | Upper Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Upper Doubler | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core (2) | | Aluminum honeycomb as given in BMS 4-4, Type 3-15NPA, Grade I. Refer to Figure 6 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-10NPA, Grade I. Refer to Figure 6 | |
| [3] | Wedge Assembly | | | LINE NUMBER 20 AND ON |
| | Upper Skin | 0.032 (0.81) | 7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7 | |
| [4] | Nose Skin | 0.063 (1.6) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [5] | Backup Plate (2) | 0.032 (0.81) | 6013-T4 sheet as given in AMS 4347 | |
| [6] | Cove Skin (4) | 0.050 (1.27) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [7] | Cove Skin (2) | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [8] | Cove Skin Assembly | | | |
| | Cove Skin | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| | Doubler | 0.160 (4.06) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [9] | Cover Plate | 0.063 (1.6) | 7075-T6 sheet as given in QQ-A-250/12 | |
| [10] | Flex Skin (3) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 8 | |
| [11] | Flex Tab (2) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9 | LINE NUMBER 1 TO 19 |
| [11] | Flex Tab (2) | | Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 10 | LINE NUMBER 20 AND ON |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



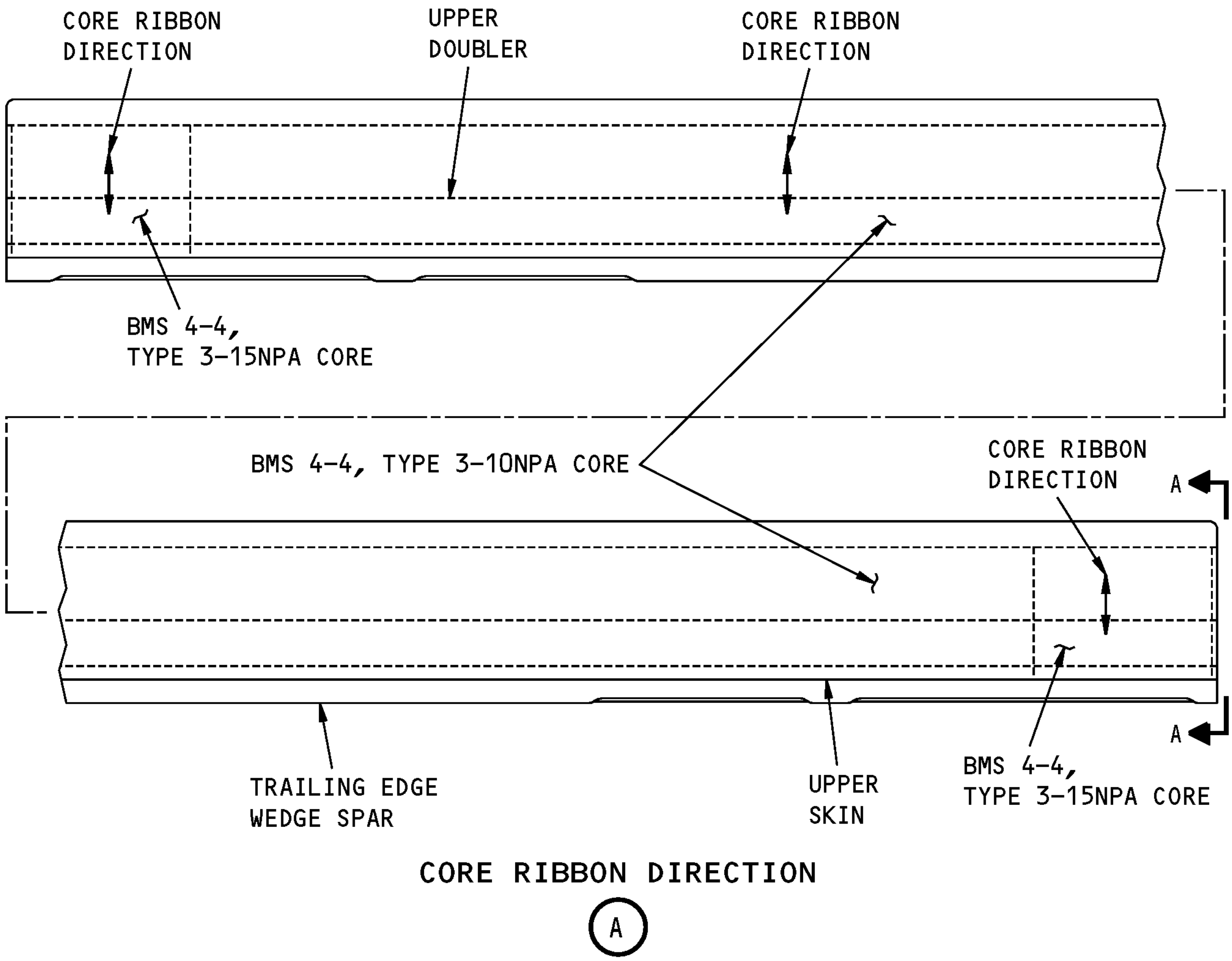

NOTE: REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

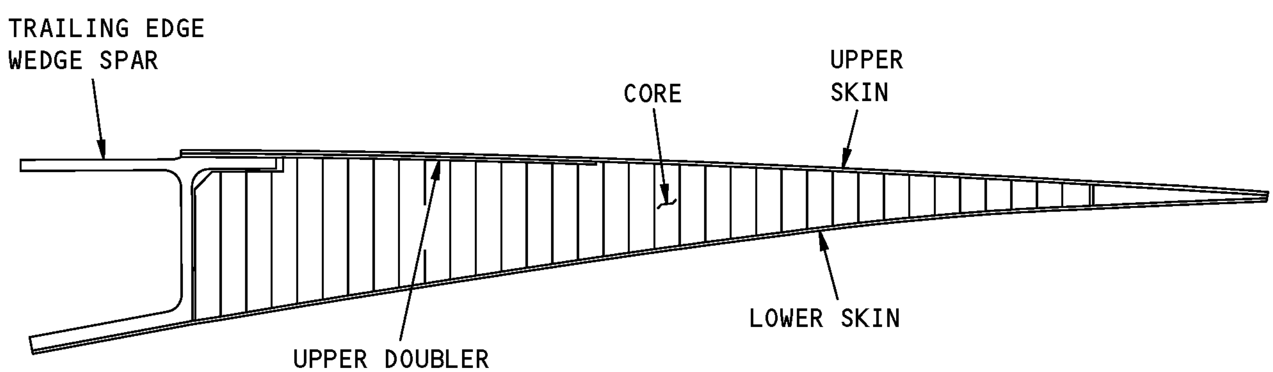
**Wing Leading Edge Slat Number 1 and Number 8 Skin Identification**
**Figure 5**

**Table 5**:

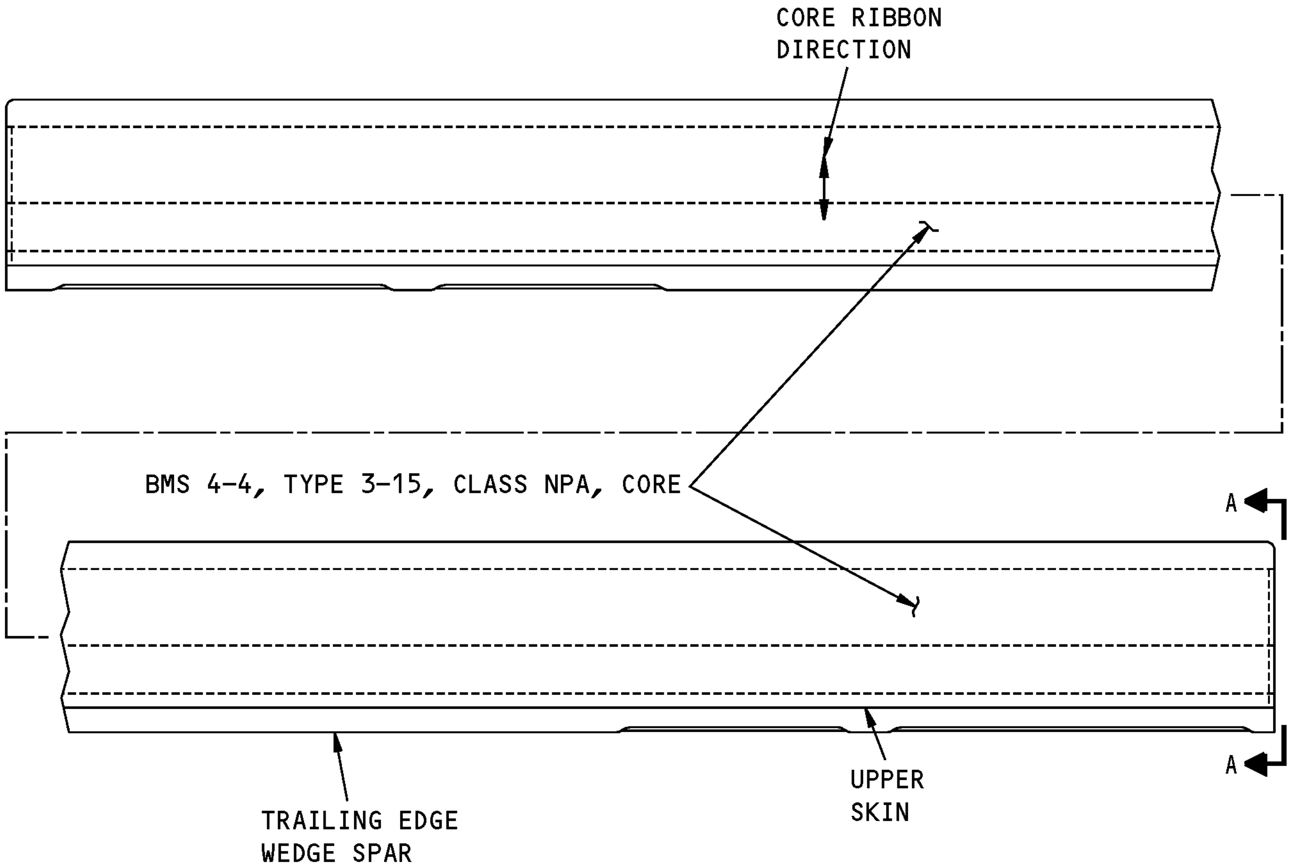
| LIST OF MATERIALS FOR FIGURE 5 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Skin Tab (2) | 0.100 (2.54) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [2] | Vortilon | | 7050-T7451 plate as given in AMS 4050 (Optional: A356.0-T6 casting as given in AMS 4218) | |
| [3] | Wedge Assembly | | | LINE NUMBER 1 TO 19 |
| | Upper Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Upper Doubler | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core (2) | | Aluminum honeycomb as given in BMS 4-4, Type 3-15NPA, Grade I. Refer to Figure 6 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-10NPA, Grade I. Refer to Figure 6 | |
| [3] | Wedge Assembly | | | LINE NUMBER 20 AND ON |
| | Upper Skin | 0.032 (0.81) | 7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7 | |
| [4] | Nose Skin | 0.063 (1.6) | 7075-T62 clad sheet as given in QQ-A-250/13 | |
| [5] | Cove Skin (4) | 0.050 (1.27) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [6] | Cove Skin (4) | 0.063 (1.6) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [7] | Flex Skin (3) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 8 | |
| [8] | Flex Tab (2) | | Epoxy impregnated glass Woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9 | LINE NUMBER 1 TO 19 |
| [8] | Flex Tab (2) | | Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 10 | LINE NUMBER 20 AND ON |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



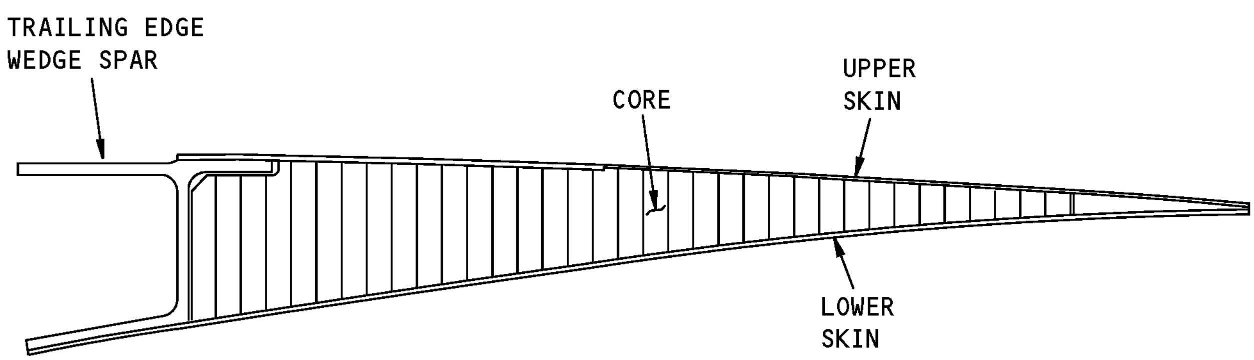




**Core Ribbon Direction for the Leading Edge Slat Wedge Assembly**
**Figure 6**



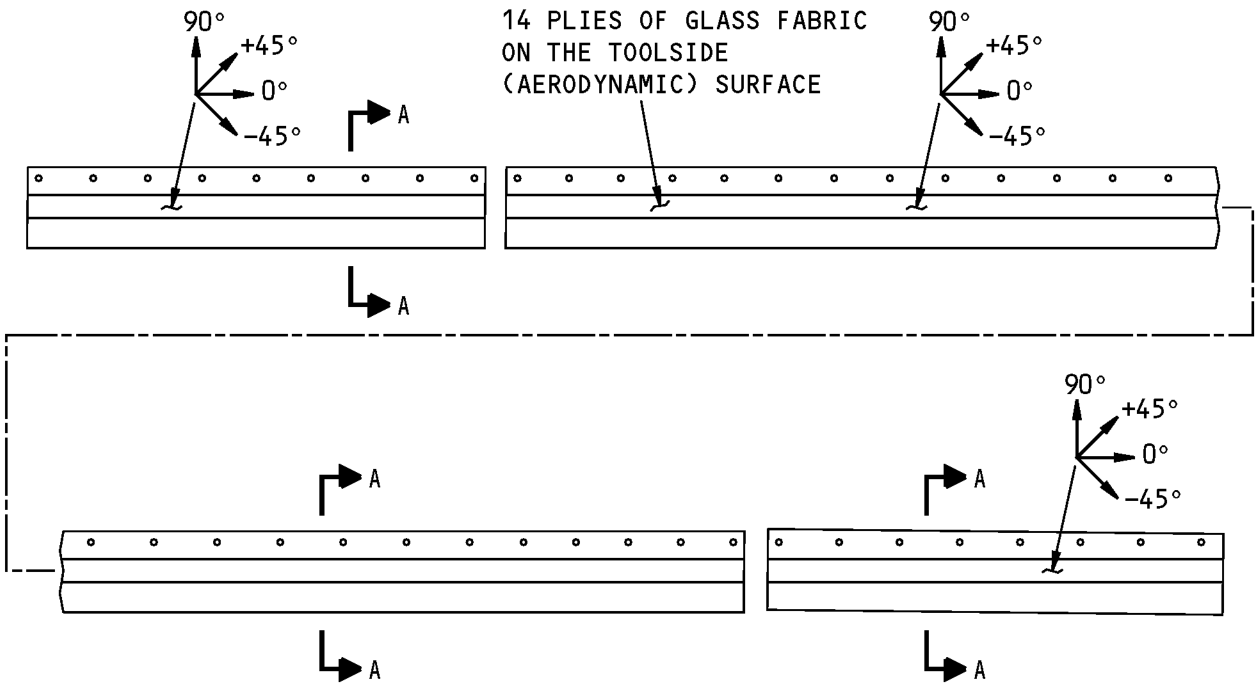

CORE RIBBON DIRECTION

A




ROTATED 90° CLOCKWISE
A-A

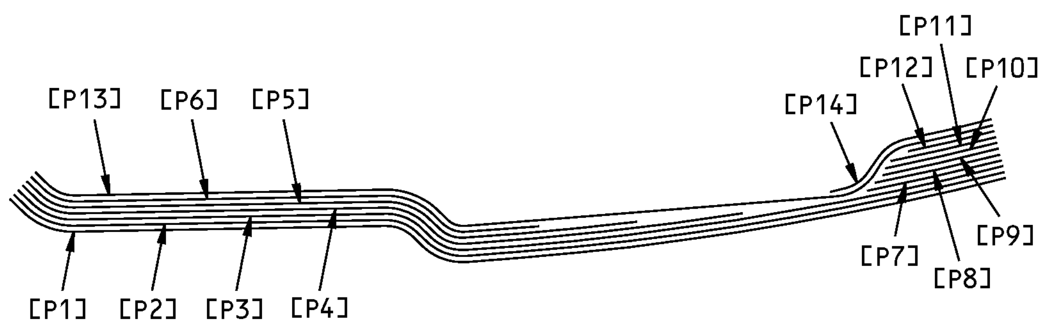
**Core Ribbon Direction for the Leading Edge Slat Wedge Assembly**
**Figure 7**




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION

A




ROTATED 90° COUNTER CLOCKWISE
PLY LAYUP SEQUENCE
A-A

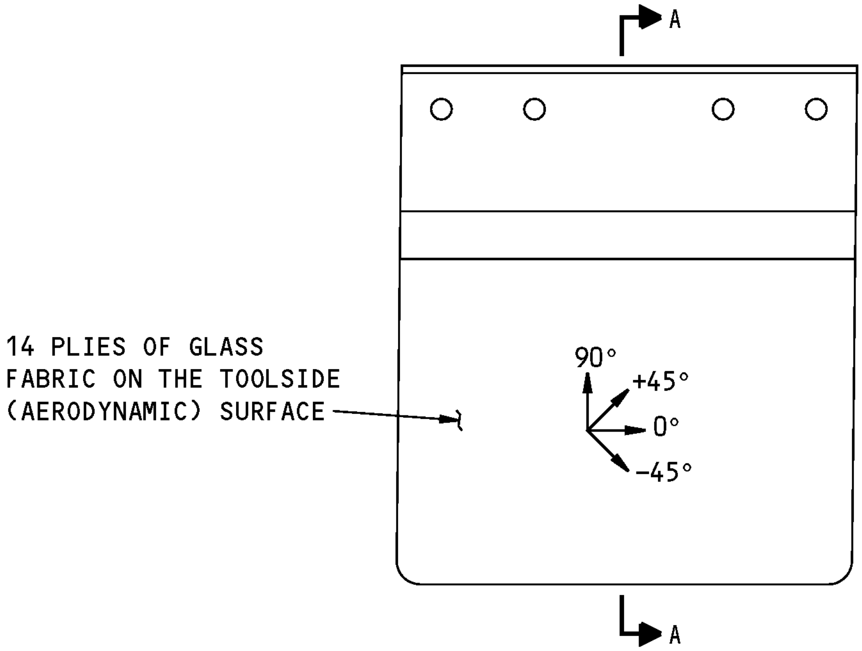
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION.
- REFER TO SECTION A-A FOR THE PLY DIRECTION AT THAT LOCATION.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction and Ply Sequence for the Leading Edge Slat Flex Skin**
**Figure 8**

**Table 6**:

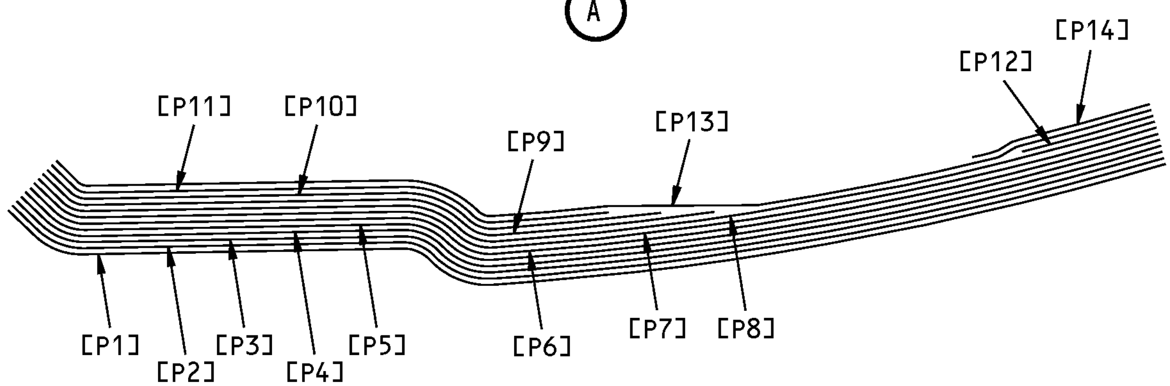
| PLY MATERIAL AND DIRECTION FOR FIGURE 8 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 thru P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION

A




ROTATED 90° COUNTER CLOCKWISE

PLY LAYUP SEQUENCE

A−A

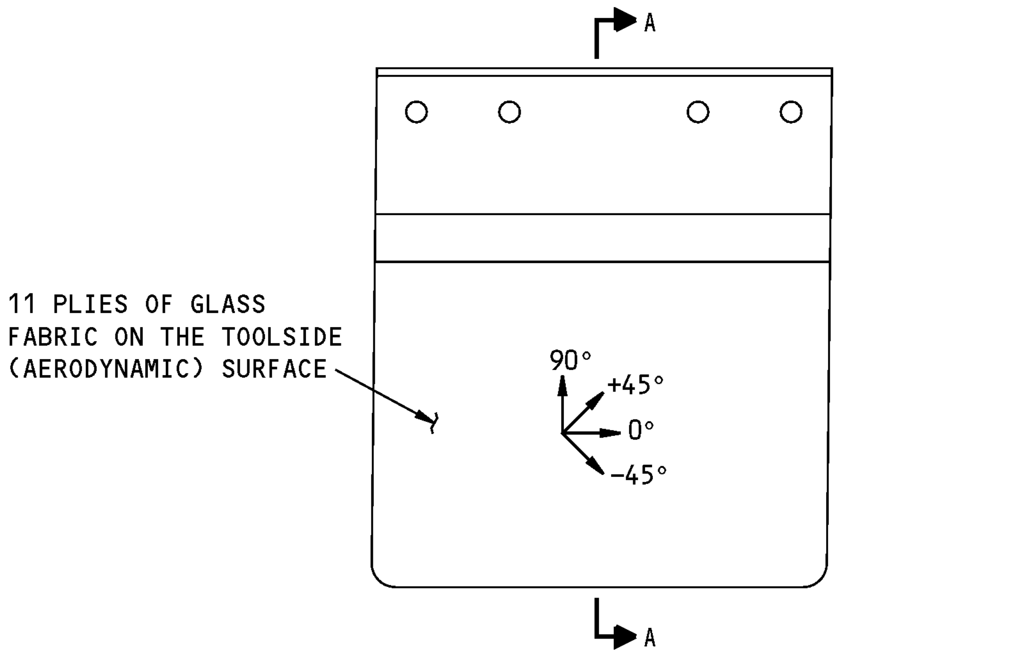
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A−A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction and Ply Sequence for the Leading Edge Slat Flex Tab**
**Figure 9**

**Table 7**:

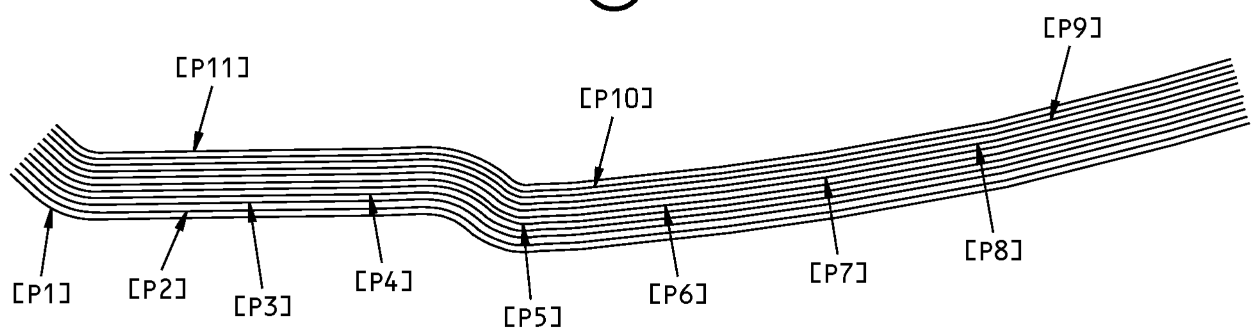
| PLY MATERIAL AND DIRECTION FOR FIGURE 9 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 thru P14 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION

A




ROTATED 90° COUNTER CLOCKWISE

PLY LAYUP SEQUENCE

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction and Ply Sequence for the Leading Edge Slat Flex Tab**
**Figure 10**

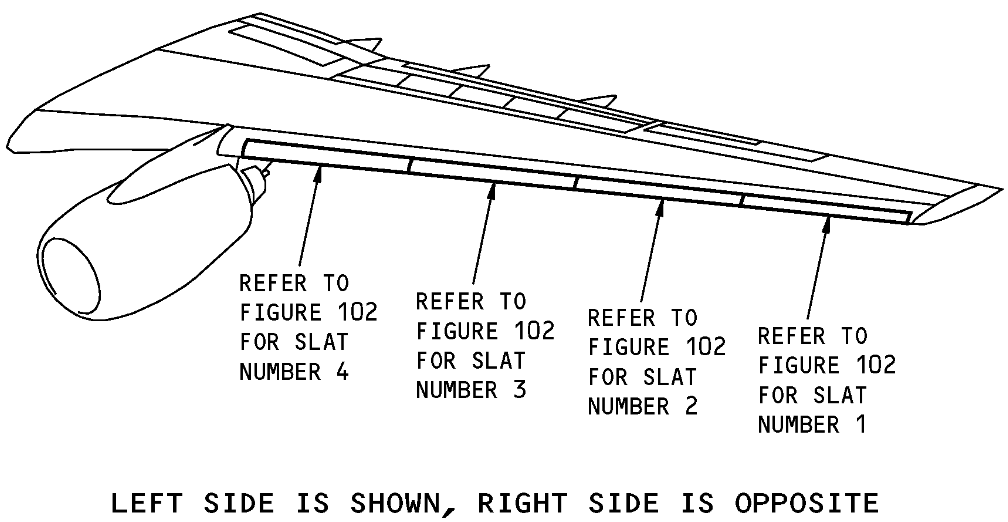
**Table 8**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 10 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 thru P11 | 0 or 90 degrees | Woven fiberglass reinforced polyetherimide as given in BMS 8–353, Type 1, Class 1, Grade 11A. |

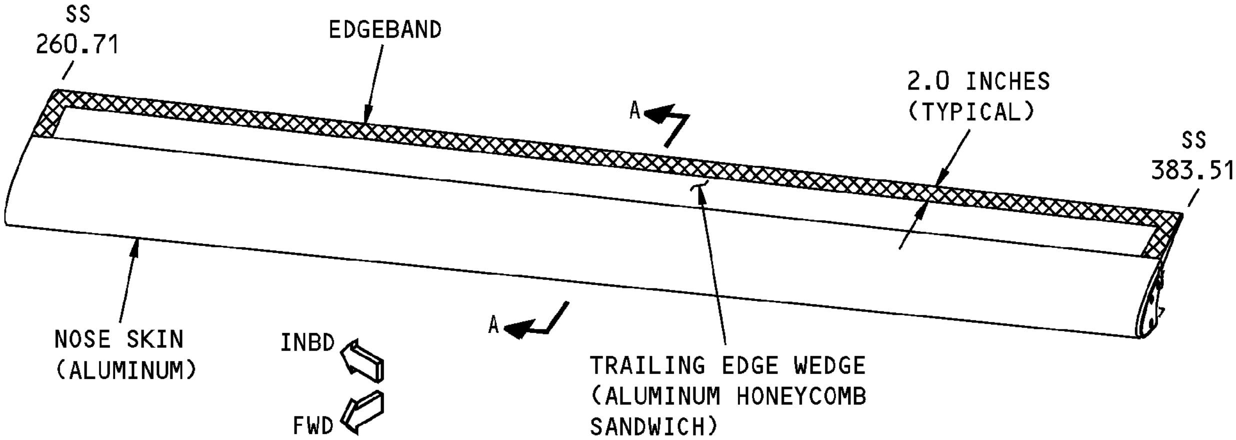
## ALLOWABLE DAMAGE 1 - WING LEADING EDGE SLAT SKIN

### 1. Applicability

A. This subject gives the allowable damage limits for the wing leading edge slat skin shown in Wing Leading Edge Slat Skin Locations, Figure 101/ALLOWABLE DAMAGE 1.

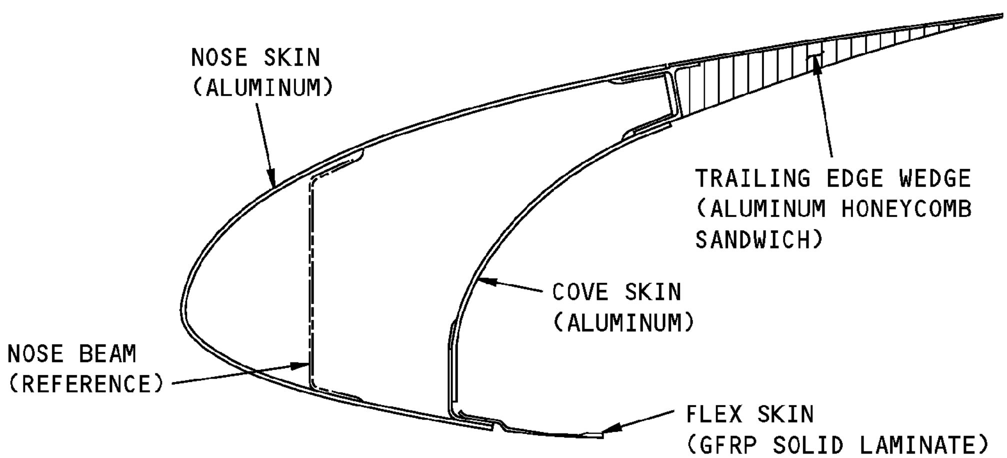



**Wing Leading Edge Slat Skin Locations**
**Figure 101**




NOTE: ALL PARTS ARE MADE OF ALUMINUM SHEET.

SLAT NUMBER 2 IS SHOWN, SLAT NUMBERS 1, 3, AND 4 ARE SIMILAR,
SLAT NUMBERS 5, 6, 7, AND 8 ARE OPPOSITE




A TYPICAL SECTION IS SHOWN
A-A

**Wing Leading Edge Slat Skin**
**Figure 102**

## 2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Do the steps that follow for the aluminum parts.

(1) Remove the damage as necessary.

(a) Refer to 51-10-02 for the inspection and removal of damage.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

(1) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(2) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.

**NOTE**: Make sure the aerodynamic smoothness on the external surfaces is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

D. Do the steps that follow for the flex skin made of Glass Fabric Reinforced Plastic (GFRP) and to the trailing edge wedge.

(1) Do an inspection of the damaged area to find the length, width, and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other equivalent inspection methods that have been examined and found to be satisfactory by the operator, can be used.

(a) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(2) Remove all the contamination and water from the structure.

(a) Refer to 51-70-04 for the damage removal procedures.

(b) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(3) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

(a) Use a vacuum and heat as necessary to remove moisture from the solid laminate. Refer to 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Make sure the tape is in satisfactory condition at normal maintenance intervals.

(e) Seal the damage permanently.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

(a) Use a vacuum and heat as necessary to remove moisture from the solid laminate. Refer to 51-70-04.

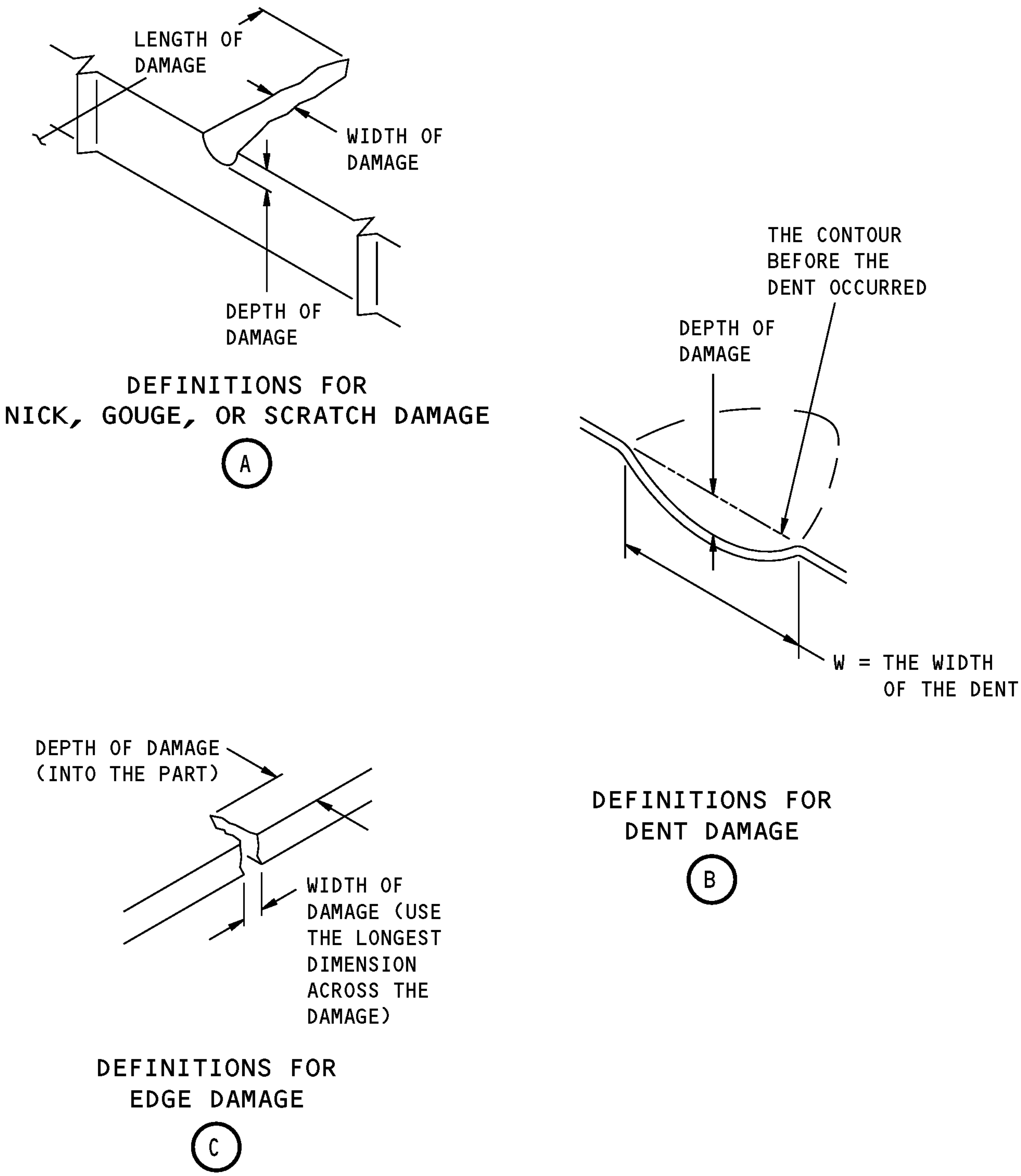
(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Seal the damage permanently no later than 24 months from the time the seal was made.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM 51-21-00/701.




**Damage Definitions**
**Figure 103**

## 3. References

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-08 | EROSION PROTECTION |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

## 4. Allowable Damage Limits

A. Nose Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail F .

1) If the damage agrees with one or more of the conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than:

- 0.10 inch aft of the nose beam, or
- 0.060 inch forward of the nose beam.

b) The distance between two dents or between a dent and other damage is less than 0.50D (D = the larger dimension of the two damaged areas).

c) There are more than 10 damage locations on a slat.

2) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval, or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 1.00 inch away from a fastener hole, material edge, or other damage

3) Filled with a 2117-T3 or 2117-T4 countersunk rivet.

a) Install the rivet without sealant.

(5) Erosion:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail I if you do as follows:

1) Blend the damage that is in the skin with a powdered household cleanser or 400 grit aluminum oxide paper.

**WARNING:** FLIGHT OPERATION IS NOT PERMITTED IF THE SLAT LEADING EDGE ROUGHNESS IS EQUIVALENT TO OR MORE THAN THAT WHICH IS MADE WITH 240 GRIT SANDPAPER.

2) You apply an erosion coating or speed tape as given in 51-20-08.

B. Trailing Edge Wedge (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Clean up cracks on the edges as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail A .

(b) Cracks are permitted in the honeycomb area if:

1) The maximum length of the damage is 1.50 inches.

2) You stop drill the ends of the crack with a 0.19 inch diameter hole. Refer to 51-10-02.

3) A minimum of 1.0 inch from a panel edge.

4) You remove water and all contaminants.

5) You make an inspection of the damaged area every 250 flight cycles, or more frequently.

6) You make a permanent repair in less than 24 months.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail G .

1) It is a maximum of 2.0 inches away from a panel edge.

2) The minimum distance away from other dent edges or other damage is 1.0 inch in diameter.

3) It is away from the wedge spar.

(b) The damage is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail H if:

1) The damage is a minimum of 2.0 inches away from the panel edge.

2) The minimum distance away from other dent edges or other damage is 1.0 inch in diameter.

(c) If the damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01

1) The depth of the dent is more than 0.120 inch and less than or equal to 0.250 inch and is as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail H .

2) The distance between two dents or between a dent and other damage is less than 1.00D (D= the larger dimension of the two damage areas).

(d) If you fill or rework the dent, do the steps that follow:

1) Do an inspection of the damage area every 250 flight cycles, or more frequently. Repair the damage in less than 24 months.

2) Repair the dent if the damage becomes larger.

3) Remove water and all contaminants.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) Cleaned up to a maximum of 0.25 inch in diameter.

2) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

(b) Remove water and all contaminants.

(c) Do an inspection of the damage every 250 flight cycles.

(d) Repair the damage in less than 24 months.

(5) Delaminations are permitted away from the edges if they are:

(a) A maximum of 2.25 inches in diameter.

(b) A minimum of 4D (D= the larger dimension of the two damaged areas) from the edge of the other damage.

(c) A minimum of 1.0 inch away from the panel edge.

(d) Inspected every 250 flight cycles, or more frequently. Repair the damage in less than 24 months.

(6) Delaminations are permitted at panel edges if they are:

(a) A maximum of 0.10 inch along the panel edge.

(b) Sealed as shown in Paragraph 2.

(c) Inspected every 250 flight cycles, or more frequently. Repair the damage in less than 24 months.

C. Cove Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E.

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail F .

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 1.00 inch away from a fastener hole, material edge, or other damage

3) Filled with a 2117-T3 or 2117-T4 countersunk rivet.

a) Install the rivet without sealant.

D. Flex Skin (GFRP Solid Laminate)

**NOTE**: Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Nicks, gouges, and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges, and scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply.

(b) A maximum of 2.0 inches in length

(c) A maximum of 0.25 inch in width

(d) A minimum of 0.50 inch away from the edge of a fastener hole

(e) A minimum distance away from the edge of other damage, a fastener hole or part edge as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 105/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.D./ALLOWABLE DAMAGE 1

(3) Dents are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A maximum of 2 plies in depth

(c) A minimum distance of 0.50 inch away from fasteners, a part edge, or other damage, a fastener hole as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 105/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.B./ALLOWABLE DAMAGE 1

(4) Holes and punctures are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A minimum distance from fasteners, a part edge, or other damage as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 105/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.D./ALLOWABLE DAMAGE 1

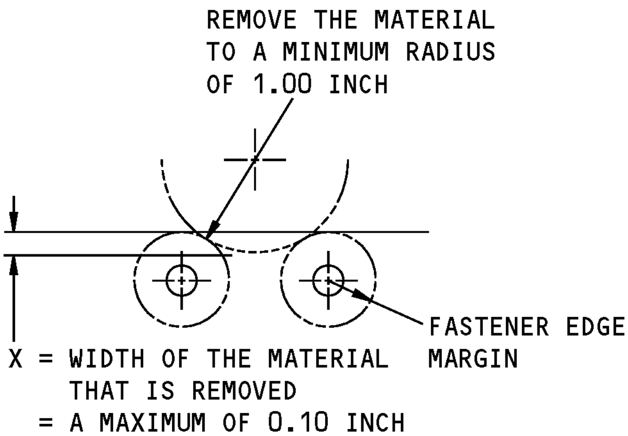
(5) Delaminations are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A minimum distance from fasteners, a part edge, or other damage as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 105/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

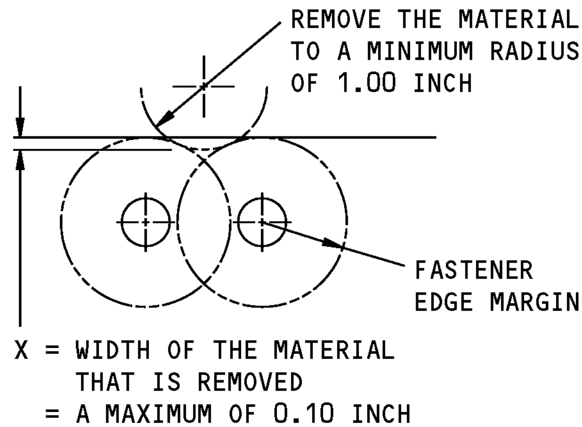
1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.B./ALLOWABLE DAMAGE 1



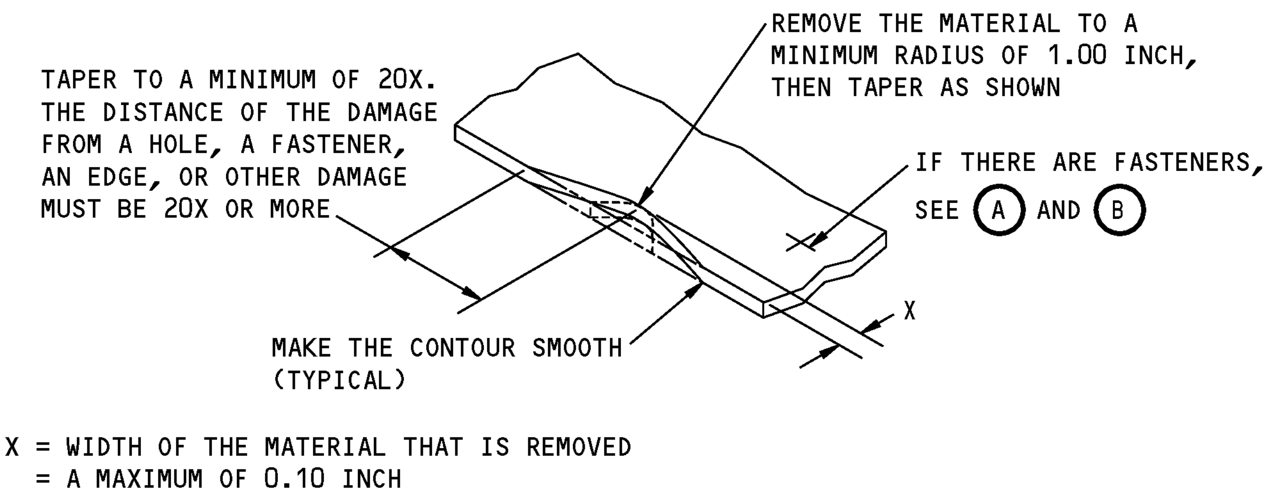

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

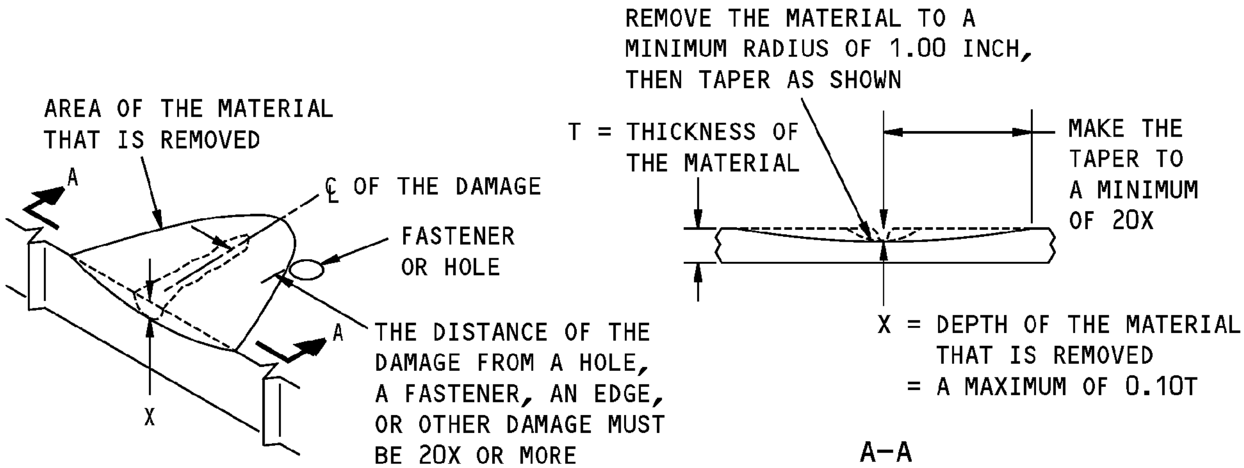
B




REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN

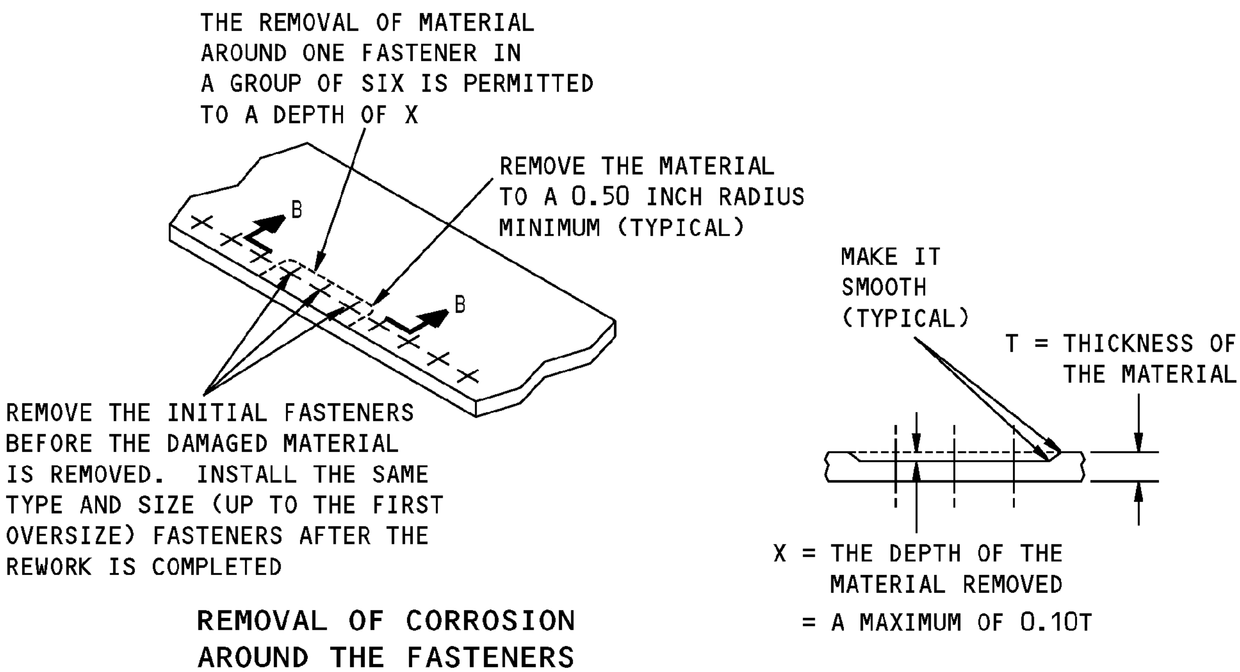
C

**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 4)**




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

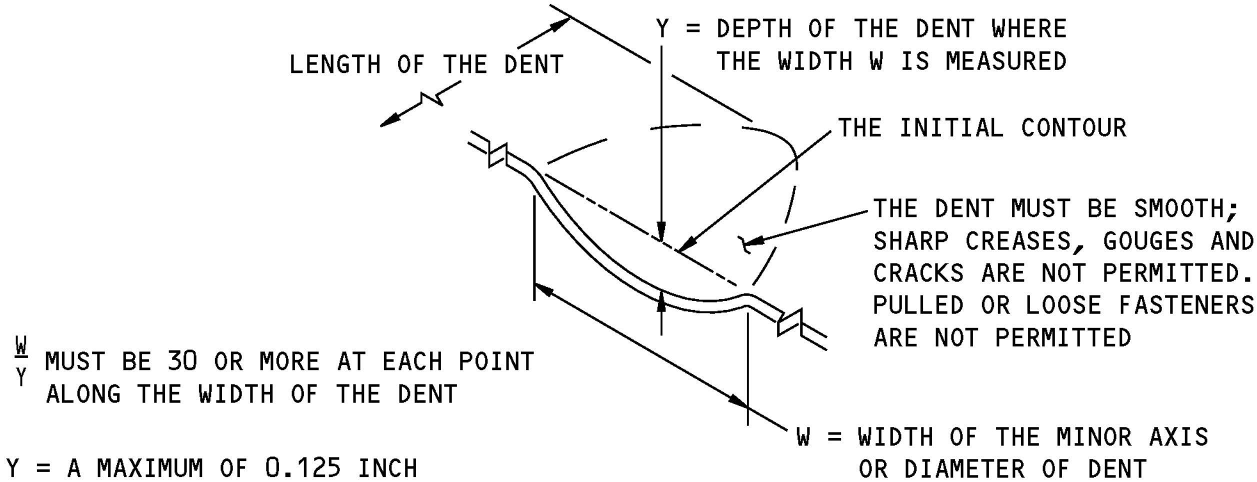
D




REMOVAL OF CORROSION
AROUND THE FASTENERS

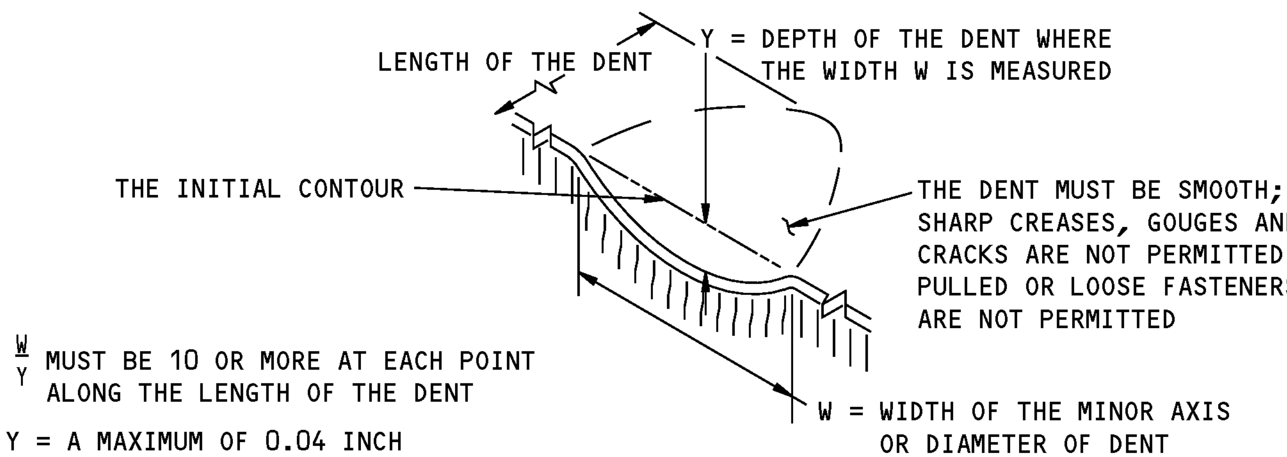
E

**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 4)**




DENT THAT IS PERMITTED IN A METAL SKIN

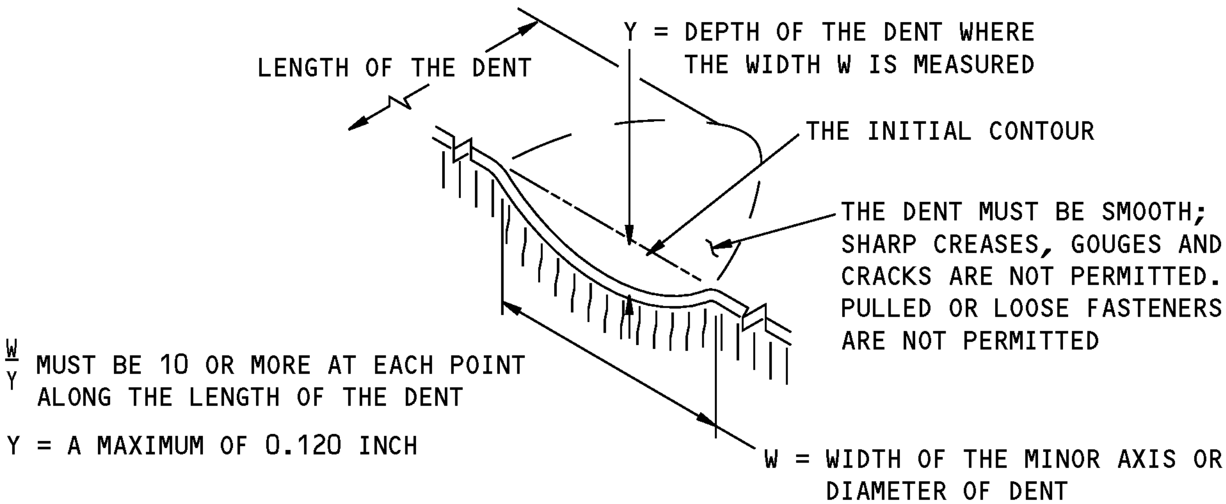
F




DENT THAT IS PERMITTED IN METAL HONEYCOMB STRUCTURE

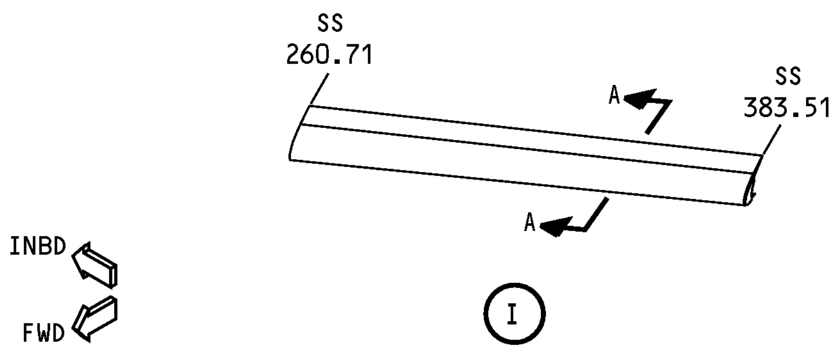
G

**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 4)**

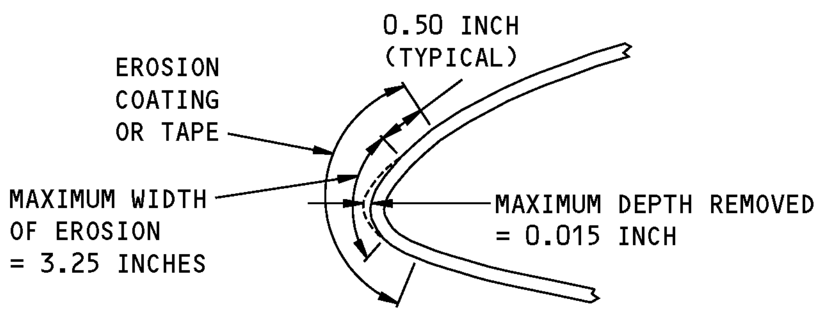



DENT THAT IS PERMITTED

H



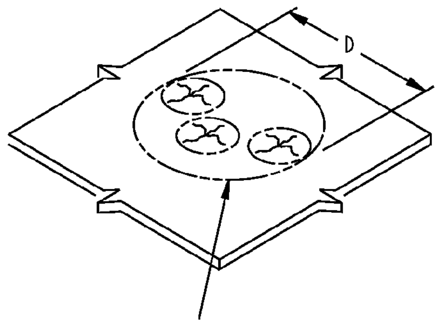

I

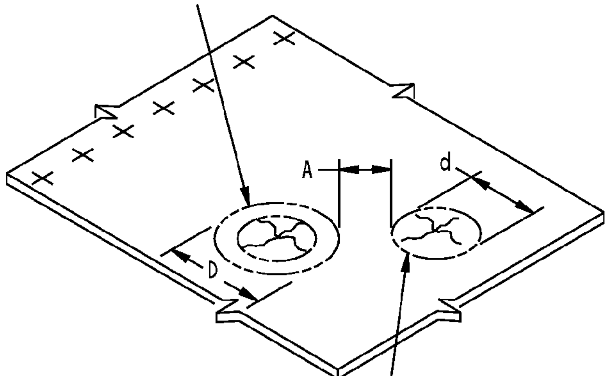



A TYPICAL SLAT LEADING EDGE SECTION IS SHOWN

A-A

**Allowable Damage Limits**
**Figure 104 (Sheet 4 of 4)**




NOTE: THE DIMENSION OF A DAMAGE AREA IS EITHER THE DIMENSION OF THE VISUAL DAMAGE OR THE DIMENSION OF THE DELAMINATION. USE THE DIMENSION OF THE LARGER DAMAGE.

D IS THE LARGER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

d IS THE SMALLER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

A IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS WHICH IS EQUIVALENT TO 1.5 X (D+2).

**Minimum Damage Spacing in Solid Laminate Parts**
**Figure 105**

## REPAIR 1 - LEADING EDGE SLAT SKIN - FLUSH REPAIR BETWEEN THE RIBS FORWARD OF THE NOSE BEAM

### 1. Applicability

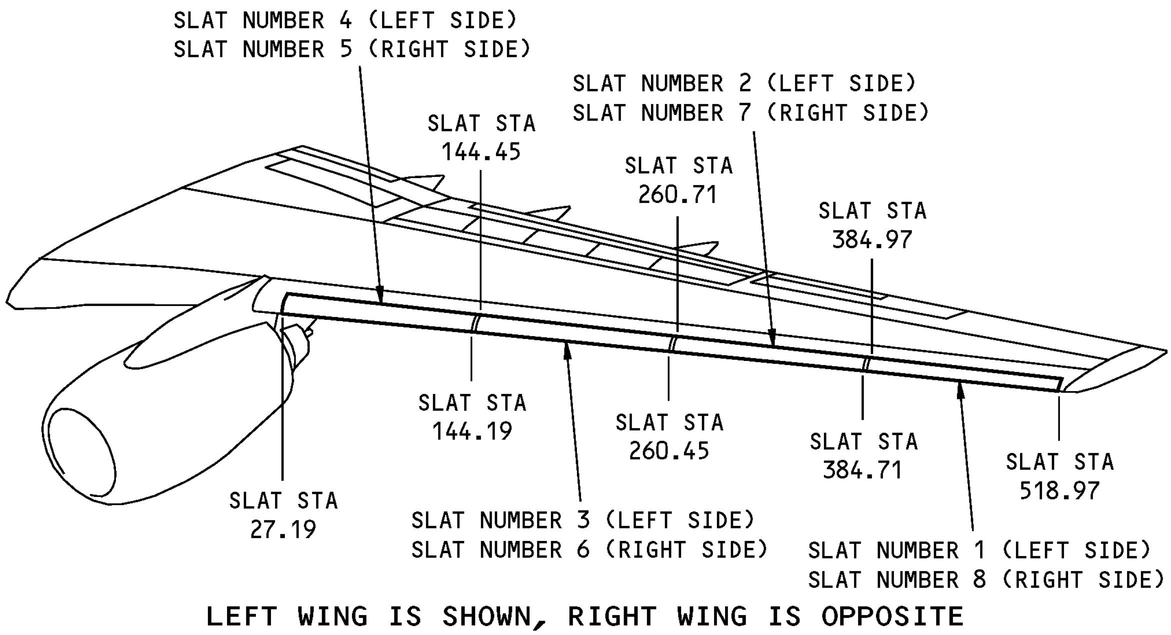
A. Repair 1 is applicable to damage that:

(1) Is on slat numbers 1 through 4 on the left wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 1

(2) Is on slat numbers 5 through 8 on the right wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 1

(3) Is on the skin forward of the nose beam and between the ribs

(4) Is not at a nose beam.




**Wing Leading Edge Slat Skins Location**
**Figure 201**

### 2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |

(Continued)

| Reference | Title |
|---|---|
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 4 | Surface Inspection of Aluminum Parts |

**4. Repair Instructions**

**CAUTION**: BE CAREFUL THAT YOU DO NOT CAUSE DAMAGE TO THE THERMAL ANTI-ICE DUCTS OF THE LEADING EDGE SLATS NUMBER 1, 2, 3, 5, 6, AND 7 WHEN YOU DO THIS REPAIR.

**CAUTION**: MAKE SURE THAT YOU DO NOT CAUSE DAMAGE TO THE STRUCTURE ADJACENT TO THE LEADING EDGE SLATS.

A. Cut and remove the damaged part of the nose skin as shown in Layout of the Repair Parts, Figure 202/REPAIR 1. Refer to 51-10-02 for the removal of damage.
   (1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to the nose beam.
   (2) Make the corner radii of the cut a minimum of 0.50 inch.

B. Put the skin around the cut back to the initial contour.

C. Do a high frequency eddy current (HFEC) inspection of the damaged area to make sure that all the damage is removed. Refer to 737 NDT Part 6, 51-00-00, Figure 4 for the HFEC inspection procedures.
   (1) The penetrant inspection is permitted as an alternative to the HFEC inspection. Refer to SOPM 20-20-02.

D. Repeat steps 4.A through 4.C until all the damage is removed. If all the damage is removed, continue with step 4.E.

E. Make the repair parts and make the shape the same as the initial contour of the skin. Refer to Table 201/REPAIR 1 and Figure 202.

F. Put the repair parts in place as shown in Layout of the Repair Parts, Figure 202/REPAIR 1.

G. Drill and countersink the necessary fastener holes.

   **NOTE**: Do not countersink more than 75 percent of the skin thickness to prevent knife-edging of the skin.

H. Remove the repair parts.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| PART NUMBER | PART | QUANTITY | MATERIAL |
| [1] | Repair Plate | 1 | Use 7075-0 clad sheet that is 0.063 inch (1.600 mm) thick. Heat treat to T62 after forming |
| [2] | Doubler | 1 | Use 7075-0 clad sheet that is 0.071 inch (1.803 mm) thick. Heat treat to T62 after forming |
| [3] | Doubler | 1 | Use 7075-0 clad sheet that is 0.071 inch (1.803 mm) thick. Heat treat to T62 after forming |
| [4] | Strap | 2 | Use 7075-T62 clad sheet that is 0.071 inch (1.803 mm) thick |

I. Remove all the nicks, scratches, burrs, gouges, and sharp edges from the repair parts and the bare surfaces of the skin.

J. Apply a chemical conversion coating to the repair parts and the bare surfaces of the initial skin. Refer to 51-20-01.

K. Apply one layer of BMS 10-11, Type I primer (Refer to SOPM 20-41-02) to:

(1) The internal and mating surfaces of the repair parts

(2) The bare surfaces of the internal surface of the skin.

L. Install the parts [2] and [3] doublers and the part [4] straps.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin, the doublers, and the straps. Refer to 51-20-05.

(2) Install the doublers and the straps through the rectangular hole.

(3) Install the BACR15CE5 fasteners without sealant. Install the NAS1739 fasteners with BMS 5-95 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

M. Install the part [1] repair plate.

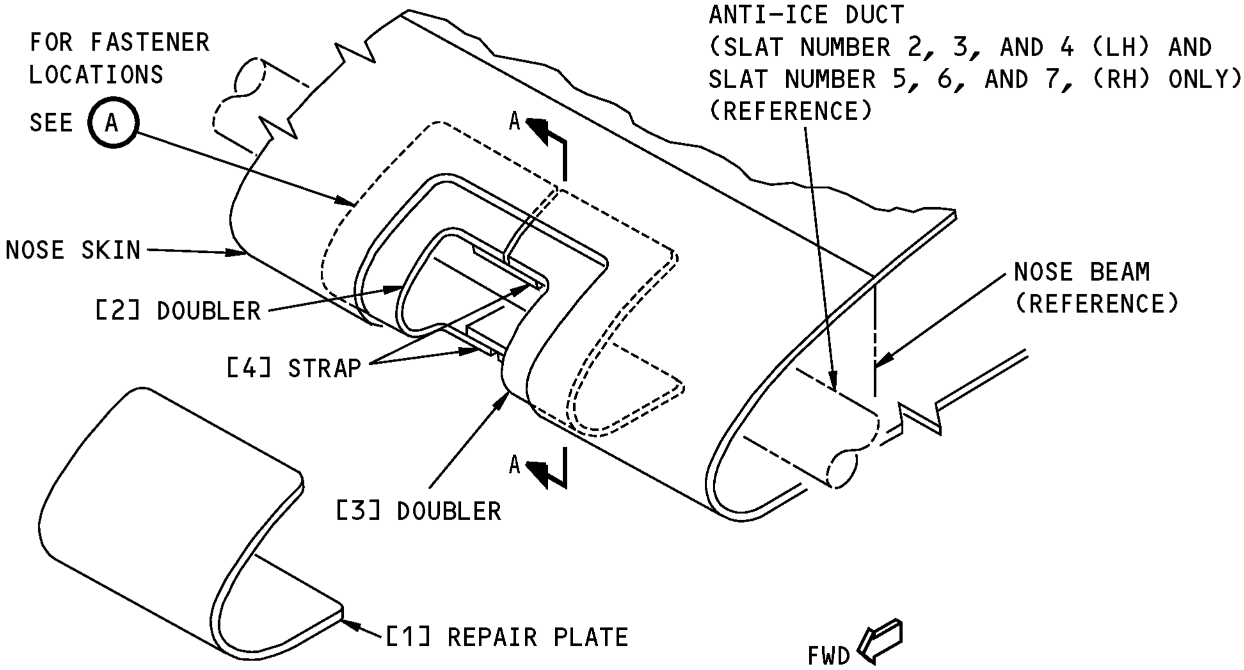
(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin, the parts [2] and [3] doublers, and the repair plate. Refer to 51-20-05.

(2) Install the repair plate.

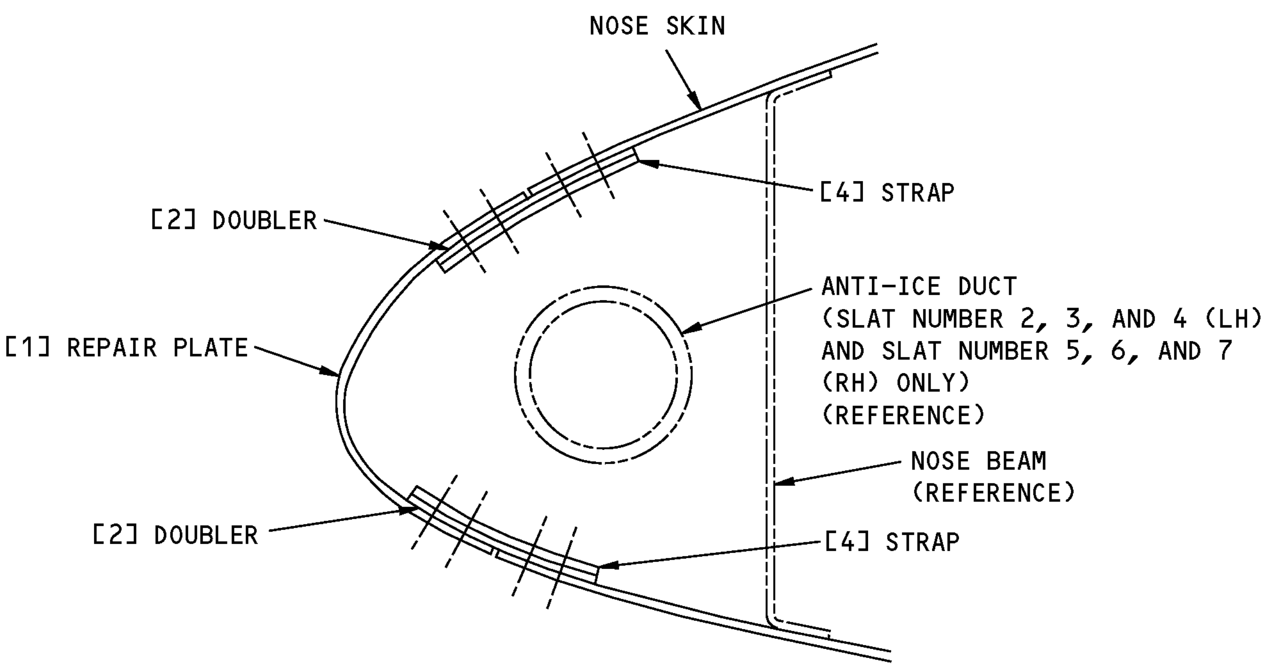
(3) Install the NAS1739 fasteners with BMS 5-95 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

N. Fill the space between the part [1] repair plate and the initial skin with BMS 5-95 sealant, if necessary. Refer to 51-20-05.

O. Apply the finish to the repair area as given in AMM 51-21-00/701.

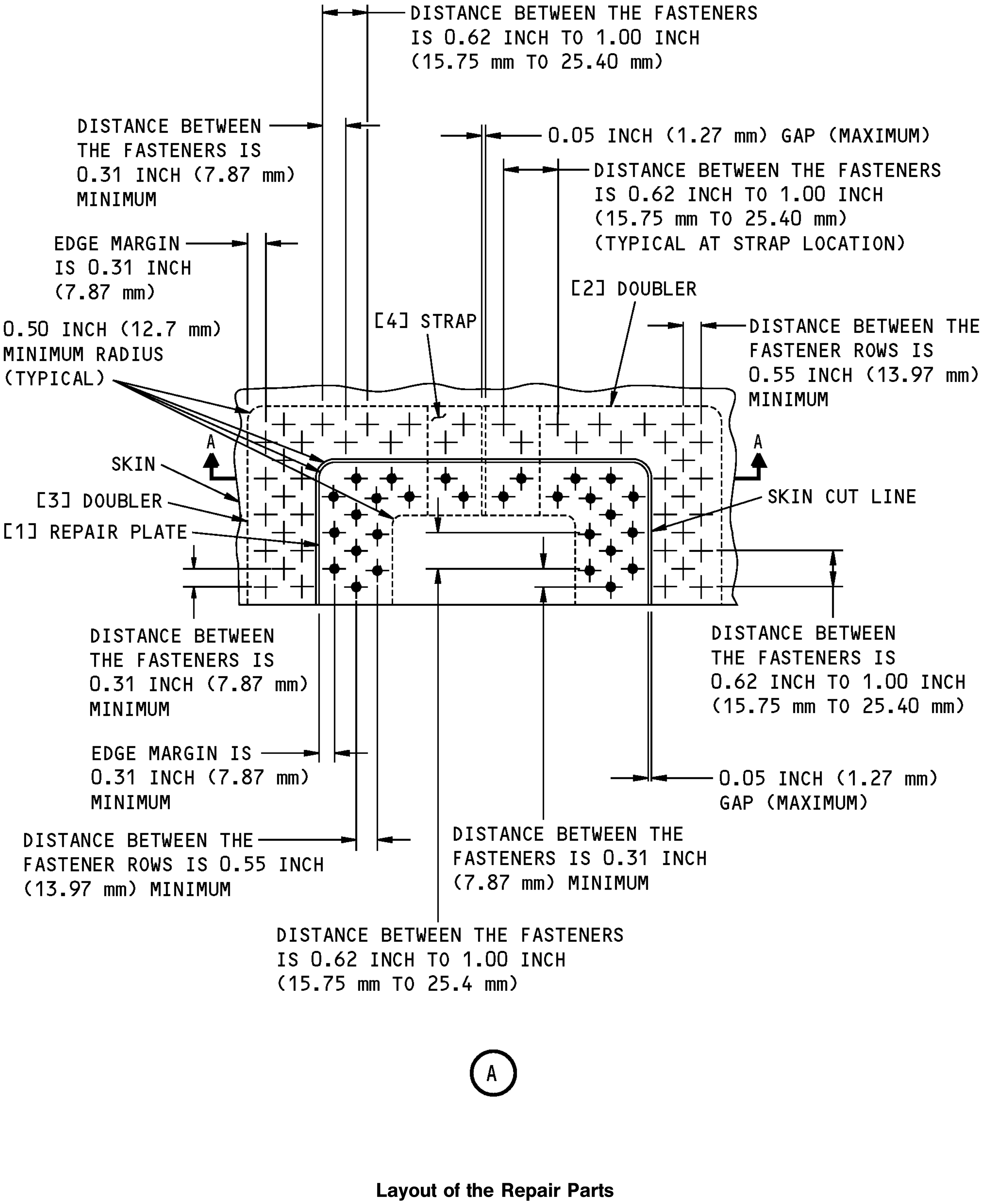



TYPICAL FOR ALL SLATS

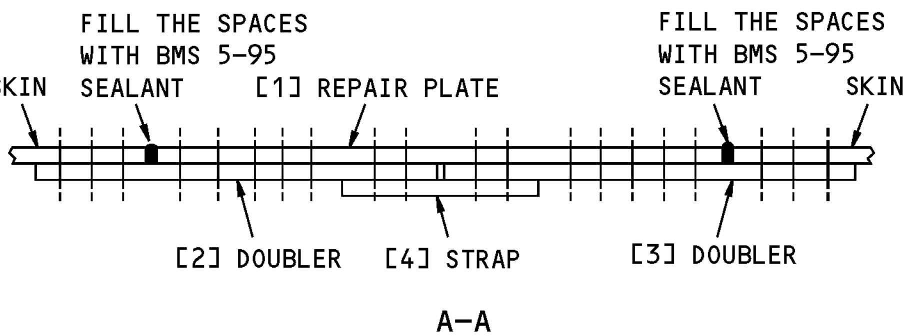



A-A

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 3)**




**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 3)**




FASTENER SYMBOLS

+ REPAIR FASTENER LOCATION. INSTALL A BACB30NW5K BOLT WITH A BACC30M COLLAR. AS AN ALTERNATIVE, INSTALL A BACR15CE5D RIVET, A NAS1739MW5, OR A NAS1739CW5 BLIND RIVET.

⌖ REPAIR FASTENER LOCATION. INSTALL A NAS1739MW5 OR NAS1739CW5 BLIND RIVET.

**Layout of the Repair Parts**
**Figure 202 (Sheet 3 of 3)**

## REPAIR 2 - LEADING EDGE SLAT SKIN - FLUSH REPAIR AFT OF THE NOSE BEAM

### 1. Applicability

A. Repair 2 is applicable to damage that:

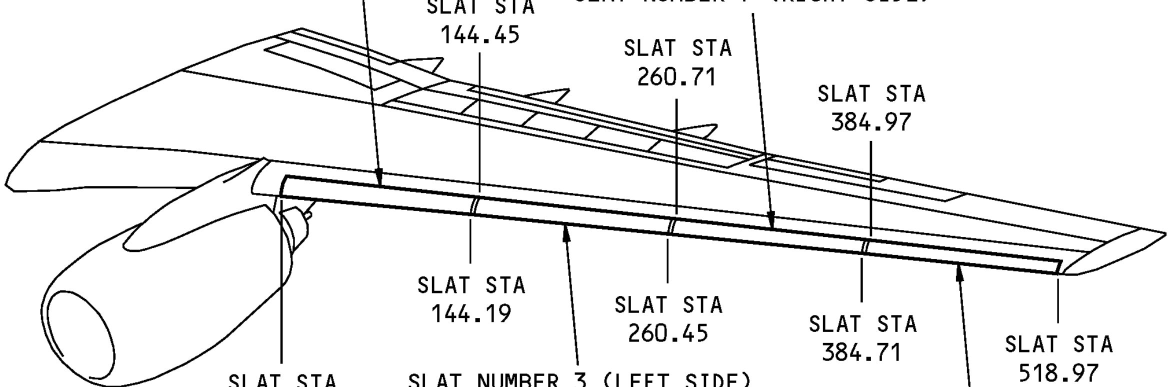
(1) Is on slat numbers 1 through 4 on the left wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 2

(2) Is on slat numbers 5 through 8 on the right wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 2

(3) Is on the skin aft of the nose beam and between the ribs

(4) Is forward of the trailing edge wedge assembly

(5) Is not at a nose beam, a cove chord, a flex skin, a main track rib, an up stop rib, an auxiliary arm rib, or an actuator rib.




**Wing Leading Edge Slat Skins Location**
**Figure 201**

### 2. General

A. Repair 2 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |

(Continued)

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 51-21-00 | INTERIOR AND EXTERIOR FINISHES |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 4 | Surface Inspection of Aluminum Parts |

## 4. Repair Instructions

**CAUTION:** MAKE SURE THAT YOU DO NOT CAUSE DAMAGE TO THE STRUCTURE ADJACENT TO THE LEADING EDGE SLATS.

A. Cut and remove the damaged part of the nose skin as shown in Layout of the Repair Parts, Figure 202/REPAIR 2. Refer to 51-10-02 for the removal of damage.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to the nose beam.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

B. Put the skin around the cut back to the initial contour.

C. Do a high frequency eddy current (HFEC) inspection of the damaged area to make sure that all the damage is removed. Refer to 737 NDT Part 6, 51-00-00, Figure 4 for the HFEC inspection procedures.

(1) The penetrant inspection is permitted as an alternative to the HFEC inspection. Refer to SOPM 20-20-02.

D. Repeat steps 4.A through 4.C until all the damage is removed. If all the damage is removed, continue with step 4.E.

E. Make the repair parts and make the shape the same as the initial contour of the skin. Refer to Table 201/REPAIR 2 and Figure 202.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| PART NUMBER | PART | QUANTITY | MATERIAL |
| [1] | Repair Plate | 1 | Use 7075-T62 clad sheet that is 0.063 inch (1.600 mm) thick |
| [2] | Doubler | 1 | Use 7075-T62 clad sheet that is 0.071 inch (1.803 mm) thick |

F. Put the repair parts in place as shown in Layout of the Repair Parts, Figure 202/REPAIR 2.

G. Drill and countersink the necessary fastener holes.

**NOTE**: Do not countersink more than 70 percent of the skin thickness to prevent a knife-edge condition in the skin.

H. Remove the repair parts.

I. Remove all the nicks, scratches, burrs, gouges, and sharp edges from the repair parts and the bare surfaces of the skin.

J. Apply a chemical conversion coating to the repair parts and the bare surfaces of the initial skin. Refer to 51-20-01.

K. Apply one layer of BMS 10-11, Type I primer (Refer to SOPM 20-41-02) to:

(1) The internal and mating surfaces of the repair parts

(2) The bare surfaces of the internal surface of the skin.

L. Install the part [2] doubler.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin and the doubler. Refer to 51-20-05.

(2) Install the doubler.

(3) Install the repair fasteners without sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 2.

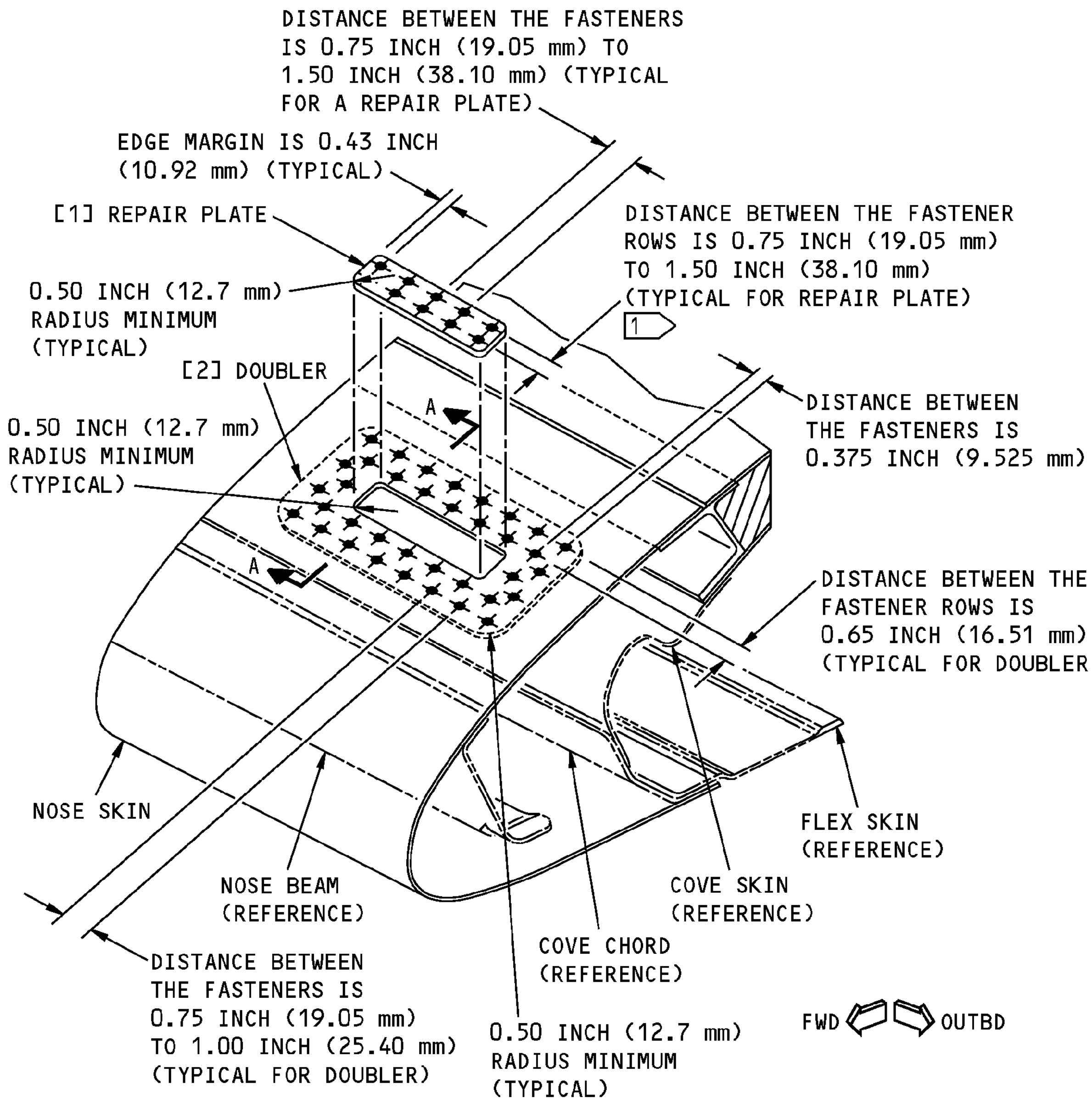
M. Install the part [1] repair plate.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin, the repair plate, and the part [2] doubler. Refer to 51-20-05.

(2) Install the repair plate.

(3) Install the repair fasteners without sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 2.

N. Fill the space between the part [1] repair plate and the initial skin with BMS 5-95 sealant, if necessary. Refer to 51-20-05.



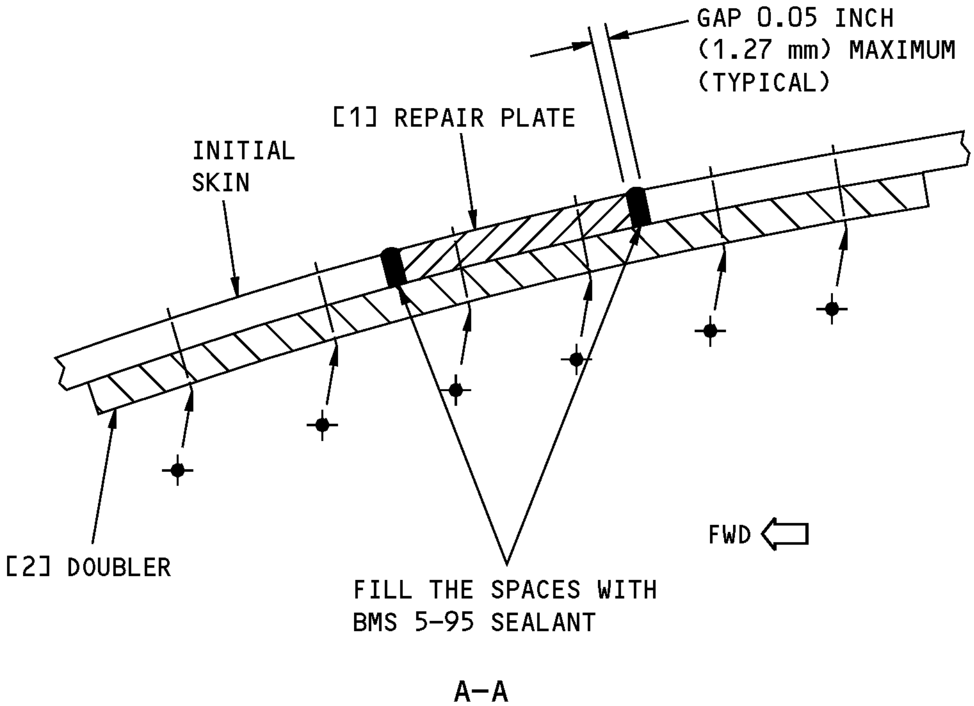

FASTENER SYMBOLS

✦ REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX DRIVE BOLT AND A BACC30M COLLAR.

NOTES

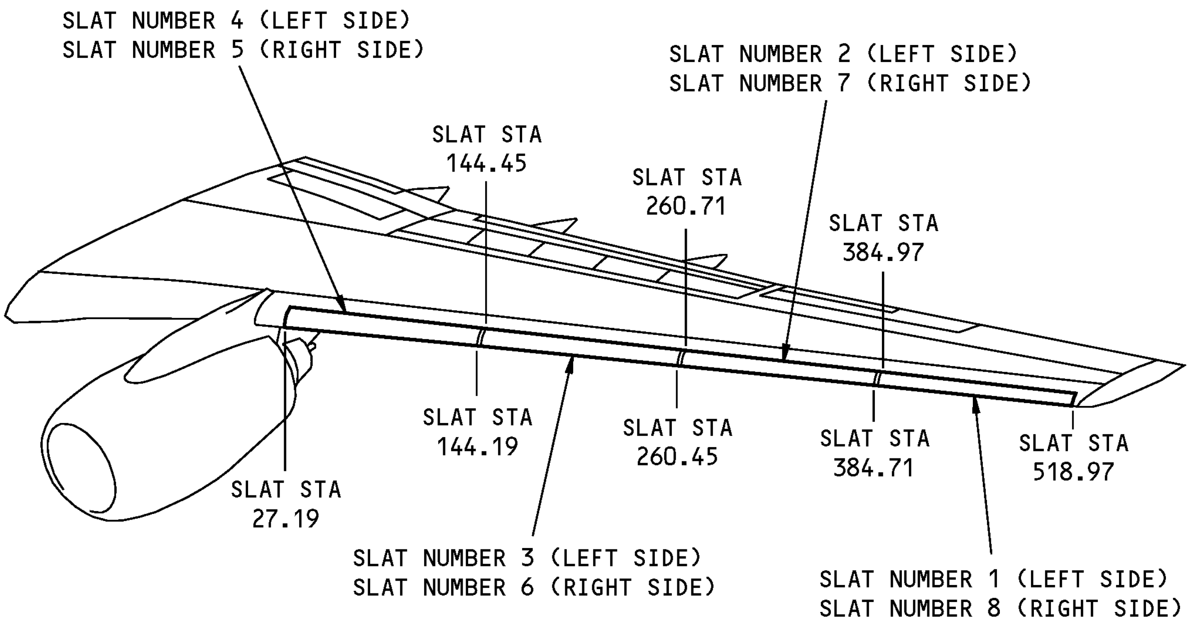
[1] YOU MAY INSTALL A MINIMUM OF ONE ROW OR A MAXIMUM NUMBER OF ROWS, AS REQUIRED, FOR THE REPAIR PLATE.

**Layout of the Repair Parts**
**Figure 202 (Sheet 1 of 2)**




**Layout of the Repair Parts**
**Figure 202 (Sheet 2 of 2)**

## REPAIR 3 - LEADING EDGE SLAT COVE SKIN

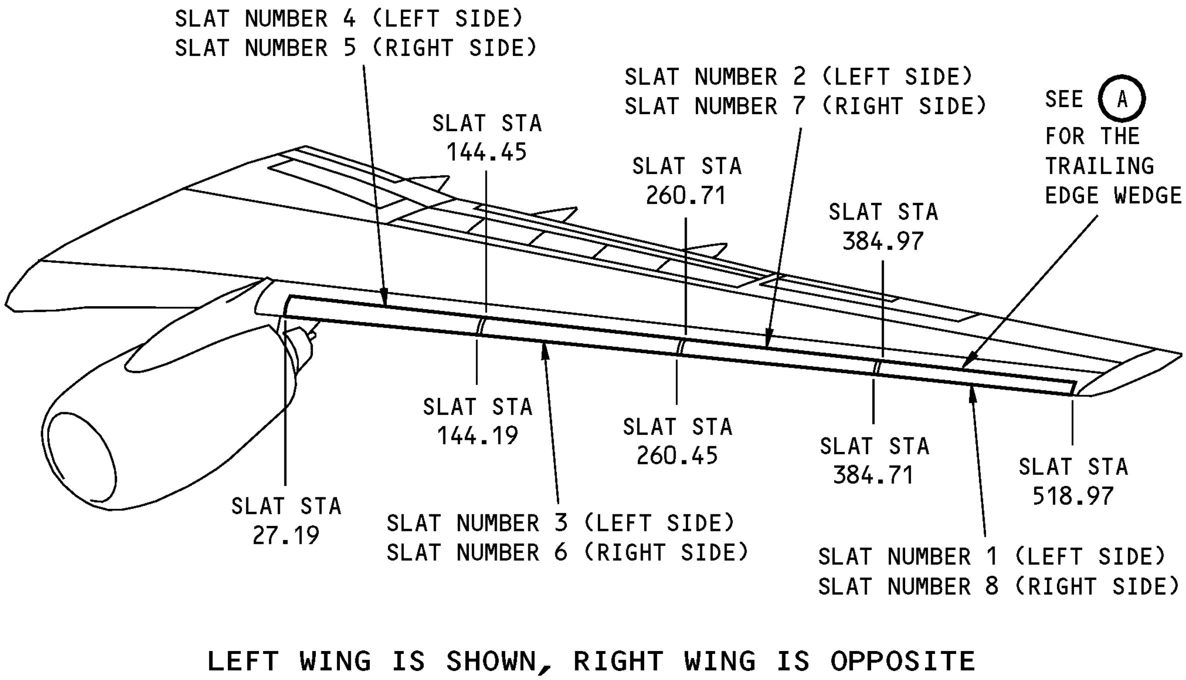



NOTE: THERE ARE NO REPAIRS FOR THE COVE SKINS OF THE WING LEADING EDGE SLATS. IF THE DAMAGE TO A COVE SKIN IS MORE THAN THE LIMITS GIVEN IN SRM 57-42-01, ALLOWABLE DAMAGE 1, REPLACE THE DAMAGED COVE SKIN.

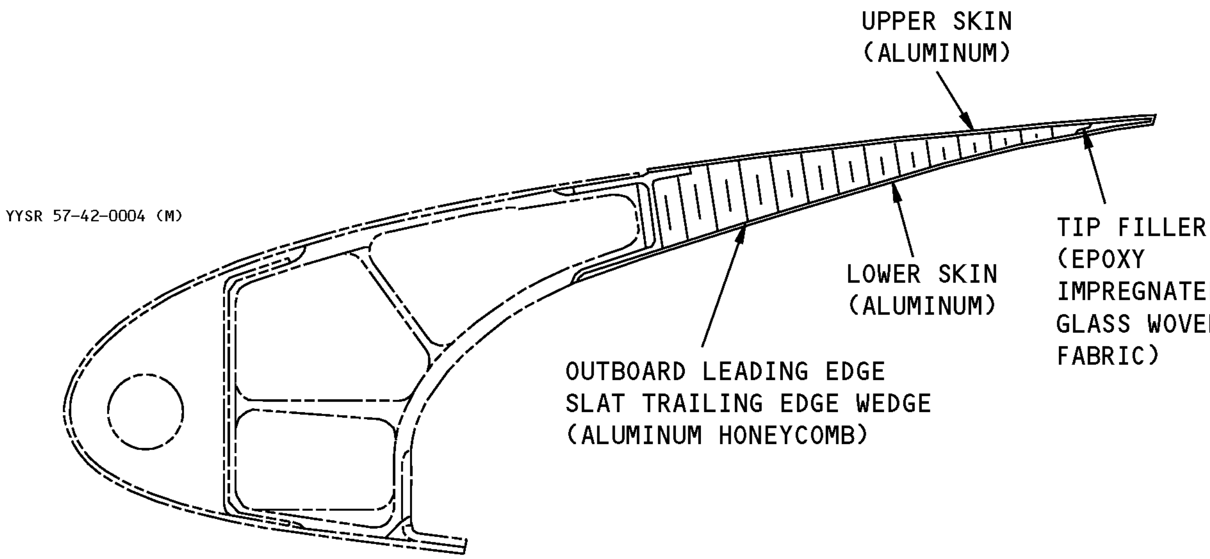
LEFT WING IS SHOWN, RIGHT WING IS OPPOSITE

**Wing Leading Edge Slat Skins Location**
**Figure 201**

## REPAIR 4 - OUTBOARD LEADING EDGE SLAT SKIN TRAILING EDGE WEDGE




**Wing Outboard Leading Edge Slat - Trailing Edge Wedge Repairs**
**Figure 201 (Sheet 1 of 2)**




A TYPICAL LEADING EDGE SLAT SECTION IS SHOWN

A

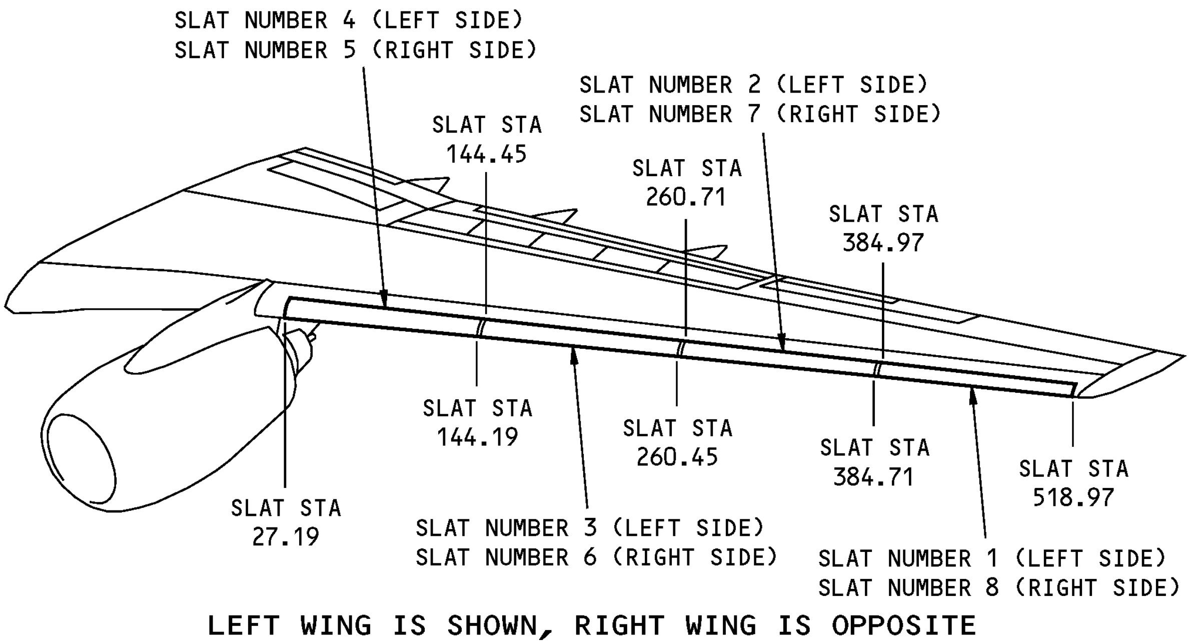
NOTE: REFER TO SRM 51-70-10 FOR TYPICAL METAL REPAIRS FOR THE OUTBOARD LEADING EDGE SLAT TRAILING EDGE WEDGE. MAKE SURE THERE IS SUFFICIENT CLEARANCE FROM ADJACENT STRUCTURE TO INSTALL THE REPAIR PARTS.

**Wing Outboard Leading Edge Slat - Trailing Edge Wedge Repairs**
**Figure 201 (Sheet 2 of 2)**

# REPAIR 5 - LEADING EDGE SLAT SKIN SPLICE REPAIR

## 1. Applicability

A. Repair 5 is applicable to damage to the wing leading edge slat skin as shown in Figure 201.




**Wing Leading Edge Slat Skins Location**
**Figure 201**

## 2. General

A. Repair 5 is a Category A repair. Refer to 51-00-06, GENERAL to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01, GENERAL.

3. **References**

| Reference | Title |
|---|---|
| 51-00-06, GENERAL | Structural Repair Definitions |
| 51-10-01, GENERAL | Aerodynamic Smoothness Requirements |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01, GENERAL | Protective Treatment of Metallic and Composite Materials |
| 51-20-05, GENERAL | Repair Sealing |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. **Repair Instructions**

A. Cut and remove the damaged part of the leading edge slat skin. Refer to 51-10-02, GENERAL for the removal of damage procedures.

(1) Make sure there is a minimum corner radius of 0.50 inch (12.70 mm) at all locations.

B. Make the repair parts as shown in Figure 202. Refer to Table 201 for the repair parts. Make sure that you form the repair parts to the contour of the initial skin.

**NOTE**: You will need to remove the initial cove skin shown in Figure 202 to permit installation of this repair.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Replacement Skin | 1 | Use clad 7075-T62 or -T6 as given in QQ-A-250/13 or as given in AMS 4049. Use the same thickness as the initial skin. |
| [2] | Tapered Filler | 8 | Use clad or bare 7075-T6. Use the same thickness as the initial skin. Make sure there is a 50:1 taper. |
| [3] | Splice Doubler | 2 | Use TI-6AL-4AV as given in Mil-T-9046 AB-1 or AMS 4911 that is 0.04 inch thick. As an alternative, you can use 17-7PH CRES as given in AMS 5528 that is 0.04 inch thick. Heat treat to 180-200 KSI. |

C. Assemble the repair parts and drill the fastener holes as shown in Figure 202.

D. Disassemble the repair parts.

E. When cutting the skin, locate the skin away from the thermal anti-ice (TAI) vent holes.

**NOTE**: If the part [1] replacement skin includes an initial thermal anti-ice (TAI) vent hole location, then make sure to back drill the thermal anti-ice (TAI) vent holes through the part [1] replacement skin to match the initial configuration.

F. It is permitted to shave the upper nose beam spacers a maximum thickness of 0.04 inch (1.02 mm) to install the part [3] splice doubler.

G. Remove all nicks, scratches, burrs, and sharp edges from the initial skin and the repair parts.

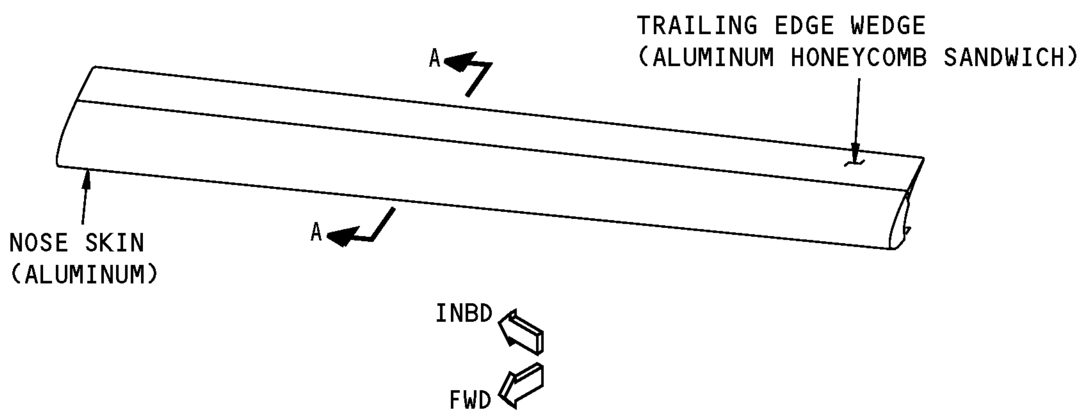
H. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial skin. Refer to 51-20-01, GENERAL.

I. Apply one layer of BMS 10-79, Type III primer to the repair parts and to the bare surfaces of the initial skin. Refer to SOPM 20-44-04.

J. Install the repair parts and fasteners wet with BMS 5-95 sealant and make sure to fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL. Install the initial cove skin to match the initial configuration

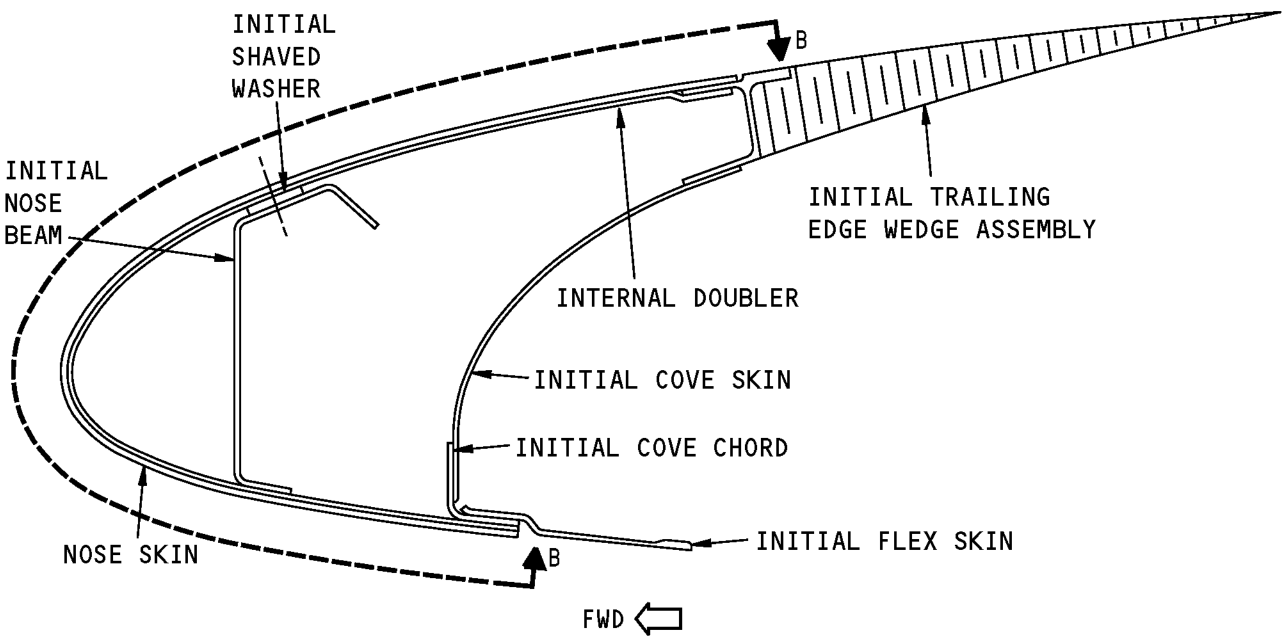
**NOTE**: Make sure there is no BMS 5-95 sealant to any of the Thermal Anti-Ice (TAI) air passages.

K. Apply a layer of BMS 10-11 Type II enamel to the repair parts and to the bare surfaces of the initial skin. Refer to SOPM 20-41-02.



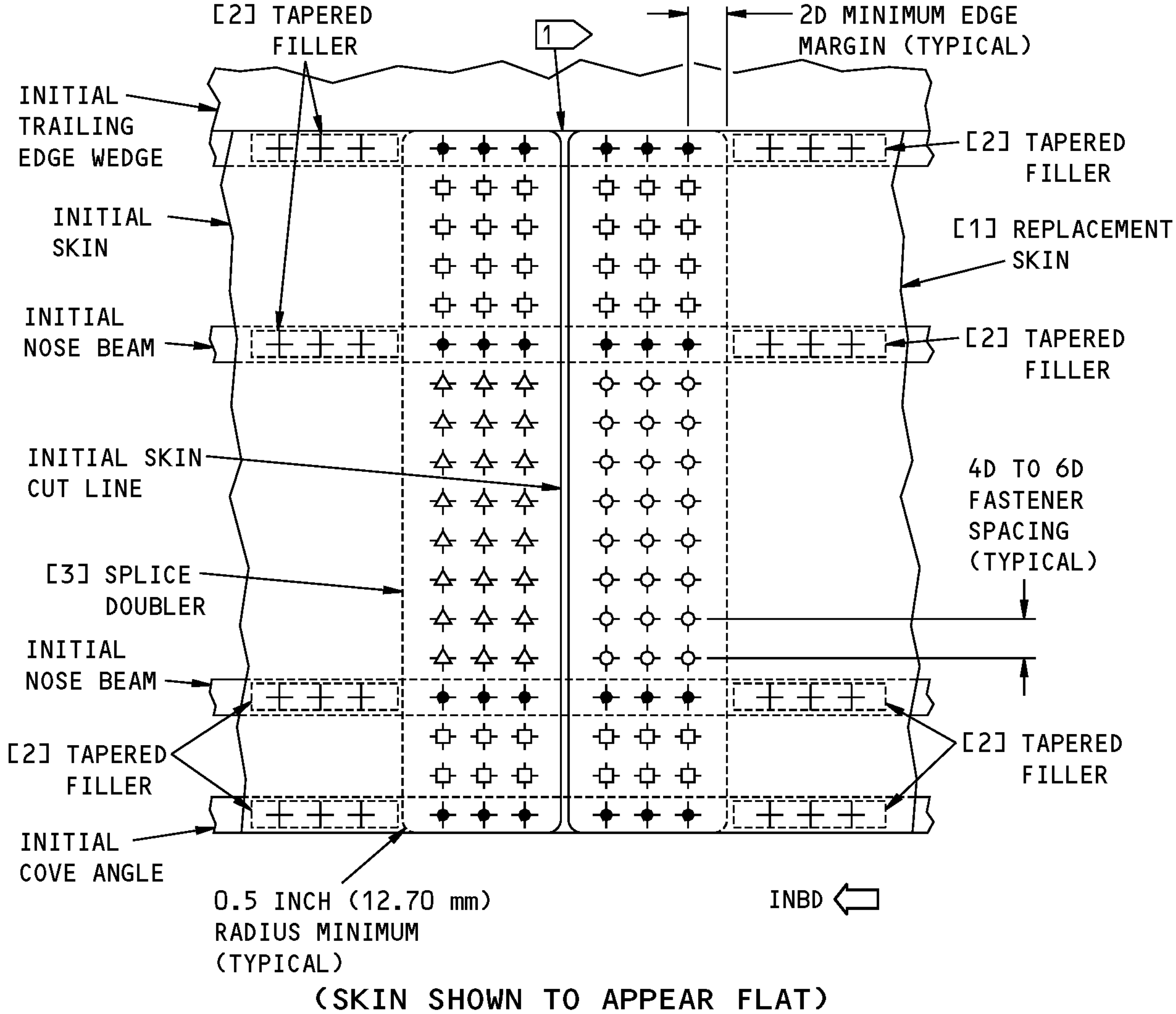

SLAT NUMBER 2 IS SHOWN, SLAT NUMBERS 1, 3, AND 4 ARE SIMILAR,
SLAT NUMBERS 5, 6, 7, AND 8 ARE OPPOSITE

A




TYPICAL SLAT CROSS SECTION
A-A

**Leading Edge Slat Skin Splice Repair**
**Figure 202 (Sheet 1 of 2)**




B-B

NOTES

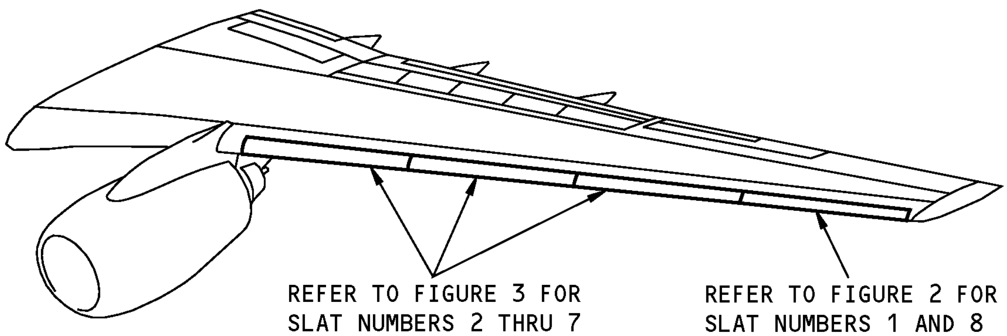
1 USE TWO SPLICES IF THE DAMAGE IS NOT NEAR THE END OF THE SLAT. USE ONE SPLICE IF THE DAMAGE IS NEAR THE END OF THE SLAT.

FASTENER SYMBOLS

- INITIAL FASTENER LOCATION. INSTALL INITIAL FASTENER TYPE
- INITIAL FASTENER LOCATION. INSTALL A BACB30NZ6Y HEX-DRIVE BOLT.
- REPAIR FASTENER LOCATION. INSTALL A PLT1058L6 BLIND RIVET AS AN ALTERNATIVE, INSTALL A PLT5470L6 OR NAS1739MW BLIND RIVET.
- REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX-DRIVE BOLT WITH A BACC30M COLLAR.
- REPAIR FASTENER LOCATION. INSTALL A BACR15CE6D RIVET.

**Leading Edge Slat Skin Splice Repair**
**Figure 202 (Sheet 2 of 2)**

## IDENTIFICATION 1 - OUTBOARD WING LEADING EDGE SLAT STRUCTURE




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

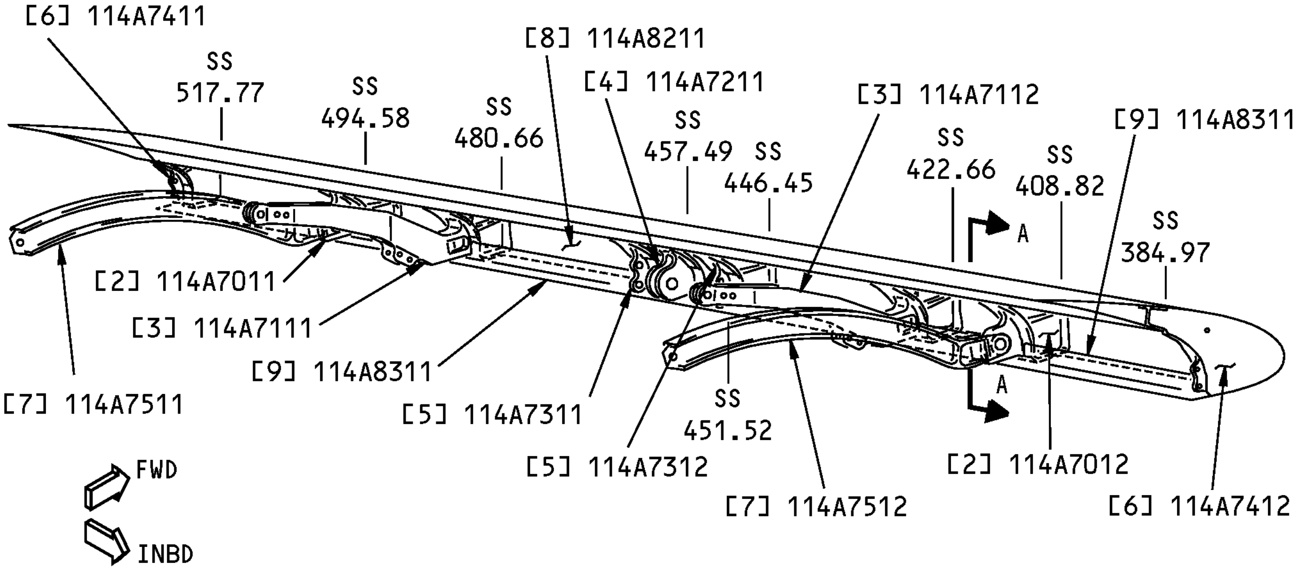
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Outboard Leading Edge Slat Structure Locations**
**Figure 1**

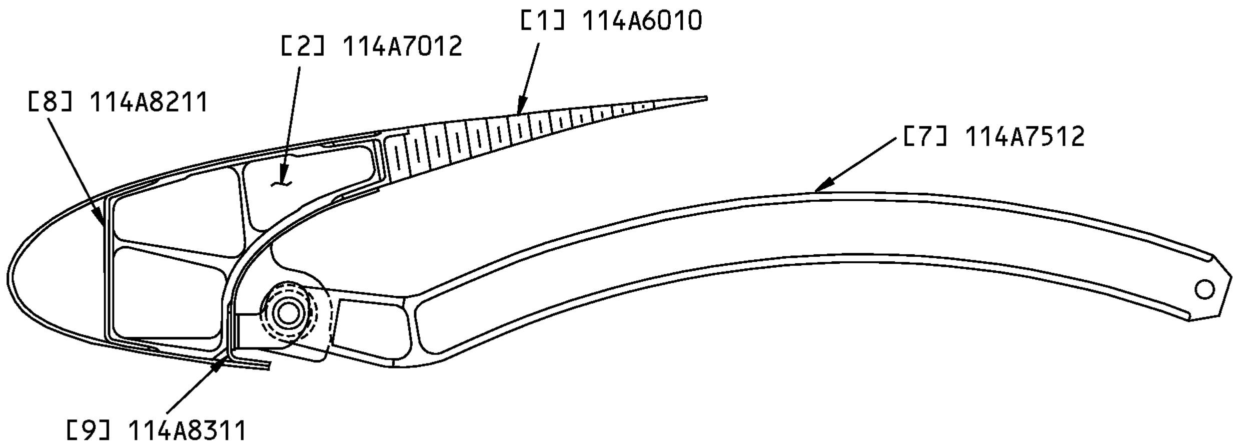
**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A1001 | Wing Left - Product Collector |
| 001A1002 | Wing Right - Product Collector |
| 114A0011 | Left Wing, Slat Functional Product Collector |
| 114A0012 | Right Wing, Slat Functional Product Collector |
| 114A4002 | Slat Installation - Outboard Leading Edge, Wing |
| 114A5010 | Slat Assembly - Number 1 and 8 |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 114A5020 | Slat Assembly - Number 2 and 7 |
| 114A5030 | Slat Assembly - Number 3 and 6 |
| 114A5040 | Slat Assembly - Number 4 and 5 |
| 114A8220 | Nose Beam Assembly - Slat 2 and 7 |
| 114A8230 | Nose Beam Assembly - Slat 3 and 6 |
| 114A8240 | Nose Beam Assembly - Slat 4 and 5 |




SLAT NUMBER 1 IS SHOWN,
SLAT NUMBER 8 IS OPPOSITE




A-A

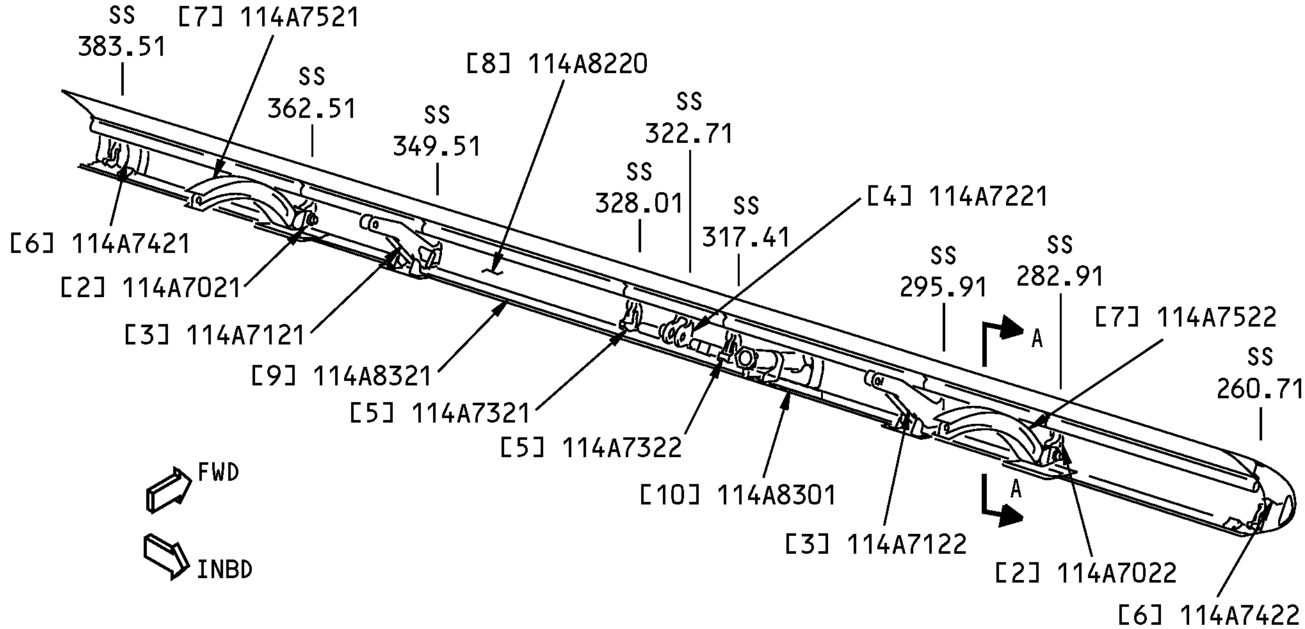
NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Wing Outboard Leading Edge Slat Structure Identification**
**Figure 2**

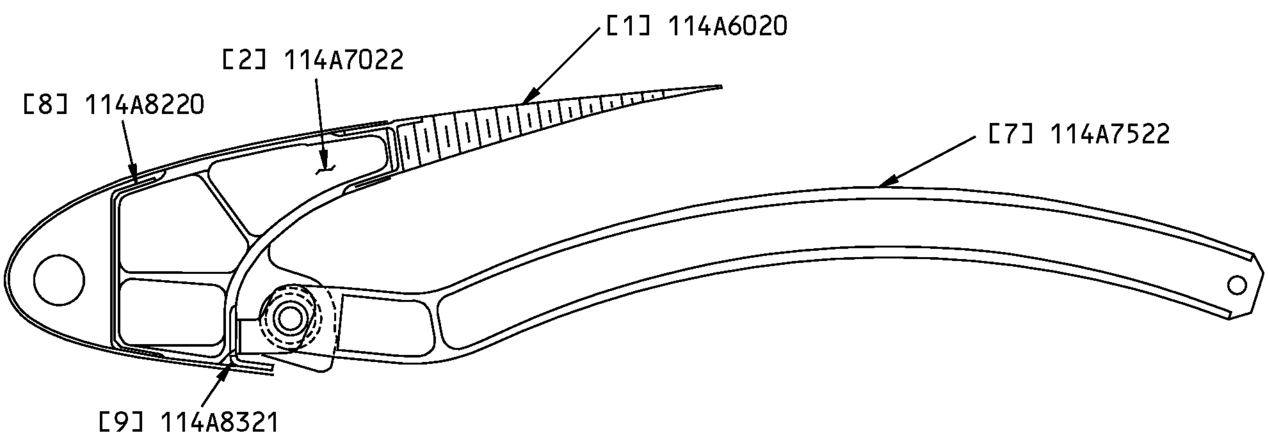
**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Wedge Assembly | | | CUM LINE 1 TO 19 |
| | Upper Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Upper Doubler | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| [2] | Main Track Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [3] | Auxiliary Arm Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part (Optional: 7050–T7451 plate as given in BMS 7-323, Type 1) | |
| [4] | Actuator Rib | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [5] | Up Stop Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [6] | End Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [7] | Main Track (2) | | 4340M forging as given in BMS 7-26. Normalized and subcritical annealed to RC 33 maximum. Heat treat to 275- 300 KSI as given in BAC 5617 | |
| [8] | Nose Beam | 0.071 (1.8) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [9] | Cove Chord | 0.080 (2.03) | 7075-T62 sheet as given in QQ-A-250/12 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




SLAT NUMBER 2 IS SHOWN,
SLAT NUMBER 7 IS OPPOSITE,
SLATS NUMBER 3 THRU 6 ARE EQUIVALENT




A-A

NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

**Wing Outboard Leading Edge Slat Structure Identification**
**Figure 3**

**Table 3**:

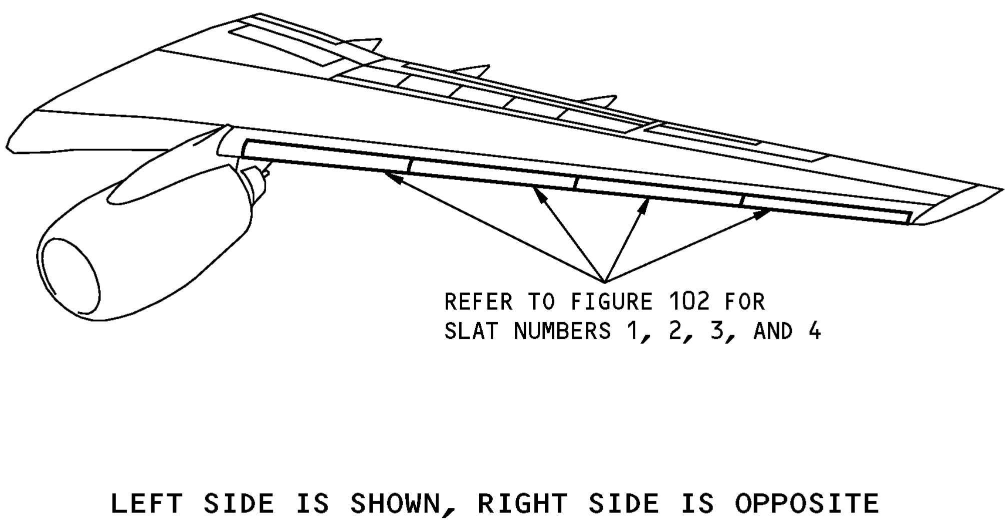
| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Wedge Assembly | | | CUM LINE 1 TO 19 |
| | Upper Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Upper Doubler | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| [1] | Wedge Assembly | | | CUM LINE 20 AND ON |
| | Upper Skin | 0.032 (0.81) | 7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses | |
| | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | Tip Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120 | |
| | Core | | Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. | |
| [2] | Main Track Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [3] | Auxiliary Arm Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part (Optional: 7050–T7451 plate as given in BMS 7-323, Type 1) | |
| [4] | Actuator Rib | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [5] | Up Stop Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [6] | End Rib (2) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |
| [7] | Main Track (2) | | 4340M forging as given in BMS 7-26. Normalized and subcritical annealed to RC 33 maximum. Heat treat to 275- 300 KSI as given in BAC 5617 | |
| [8] | Nose Beam Assembly | | | |
| | Nose Beam | 0.071 (1.8) | 7075-T62 sheet as given in QQ-A-250/12 | |
| | Doubler (2) | 0.090 (2.29) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [9] | Cove Chord | 0.080 (2.03) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [10] | Cove Chord Doubler | 0.080 (2.03) | 7075-T6 sheet as given in QQ-A-250/12 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

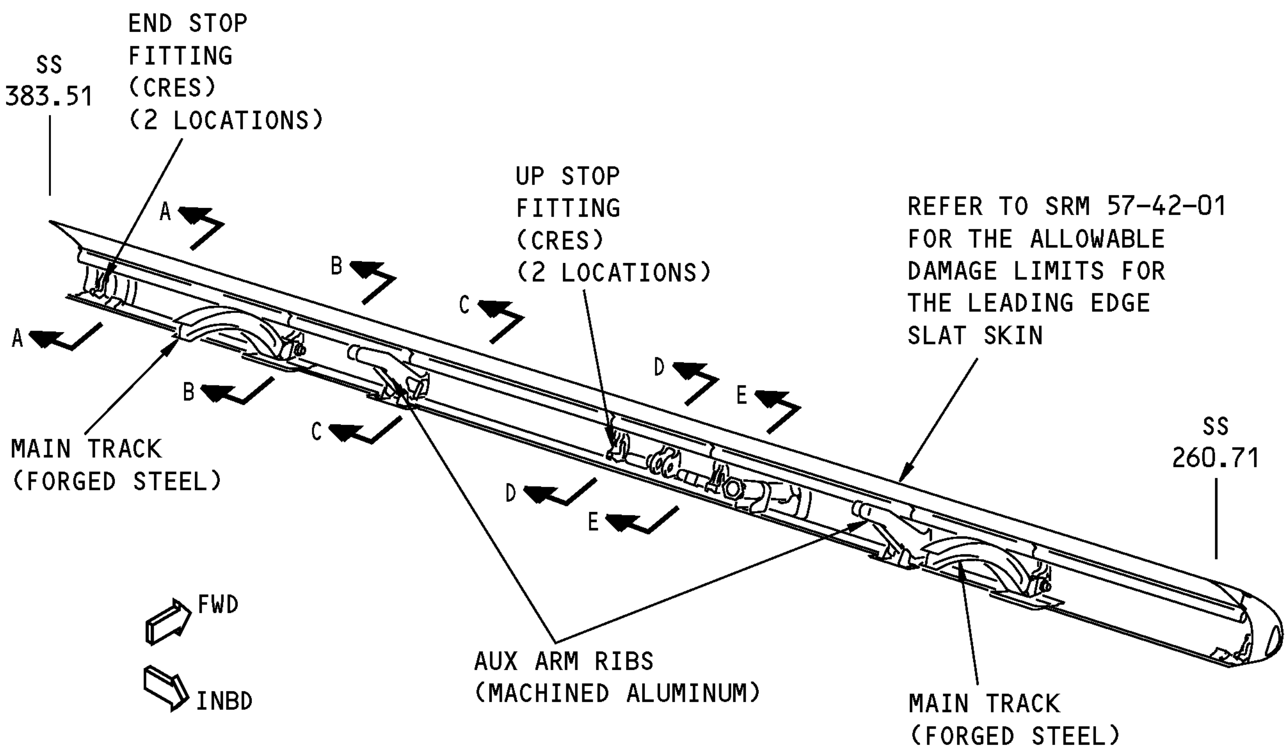
## ALLOWABLE DAMAGE 1 - WING LEADING EDGE SLAT STRUCTURE

1. **Applicability**

   A. This subject gives the allowable damage limits for the wing leading edge slat structure shown in Wing Outboard Leading Edge Slat Structure Locations, Figure 101/ALLOWABLE DAMAGE 1.

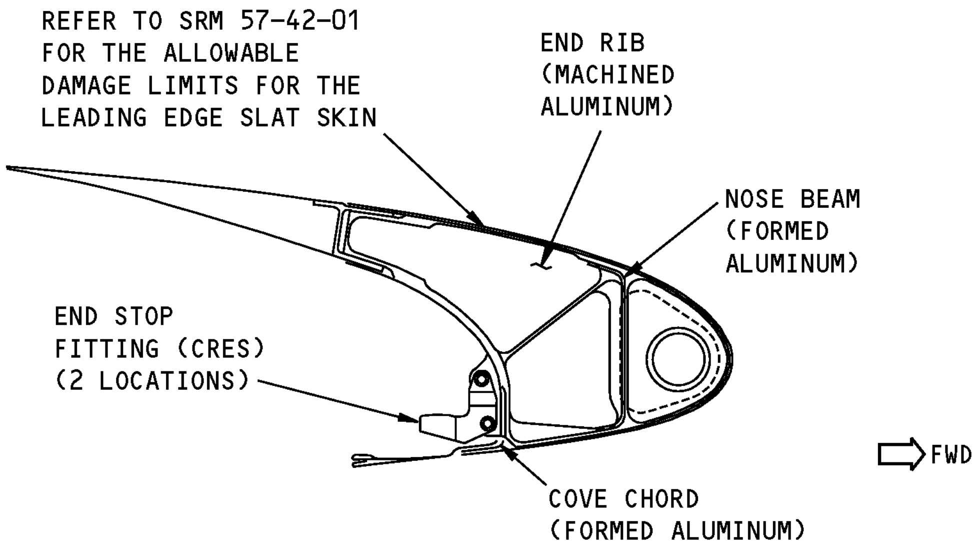



**Wing Outboard Leading Edge Slat Structure Locations**
**Figure 101**

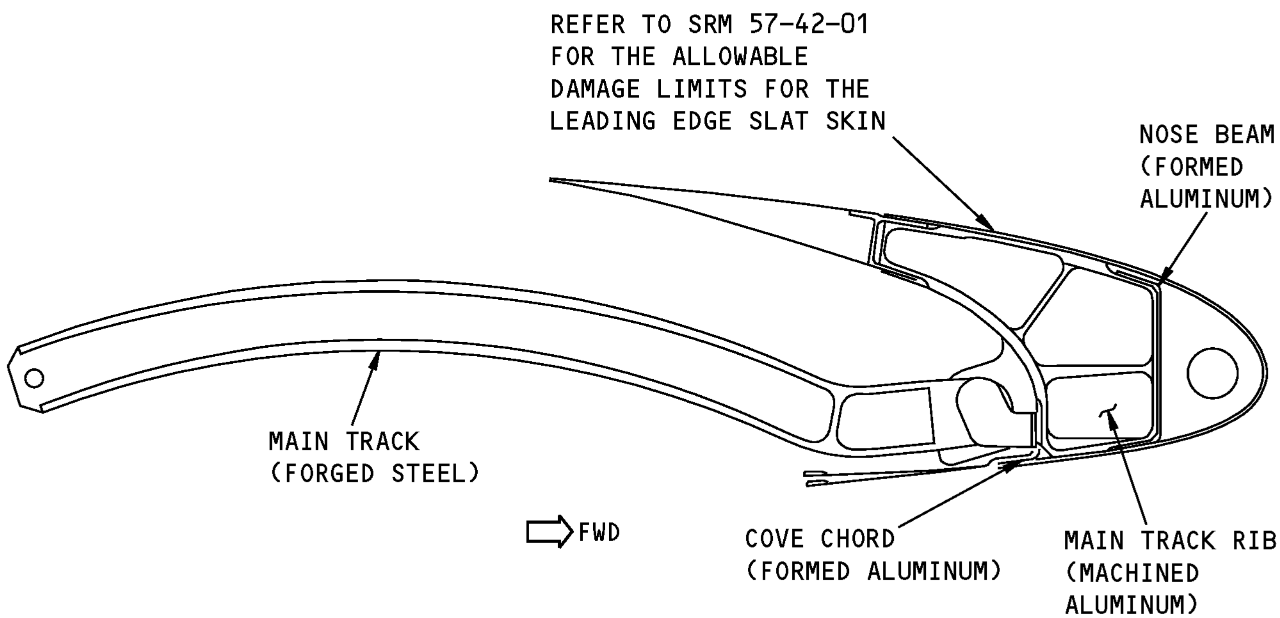



SLAT NUMBER 2 IS SHOWN, (ROTATED 180°)
SLATS NUMBER 1, 3, AND 4 ARE SIMILAR
SLAT NUMBER 8 IS OPPOSITE

**Outboard Leading Edge Slat Structure**
**Figure 102 (Sheet 1 of 3)**

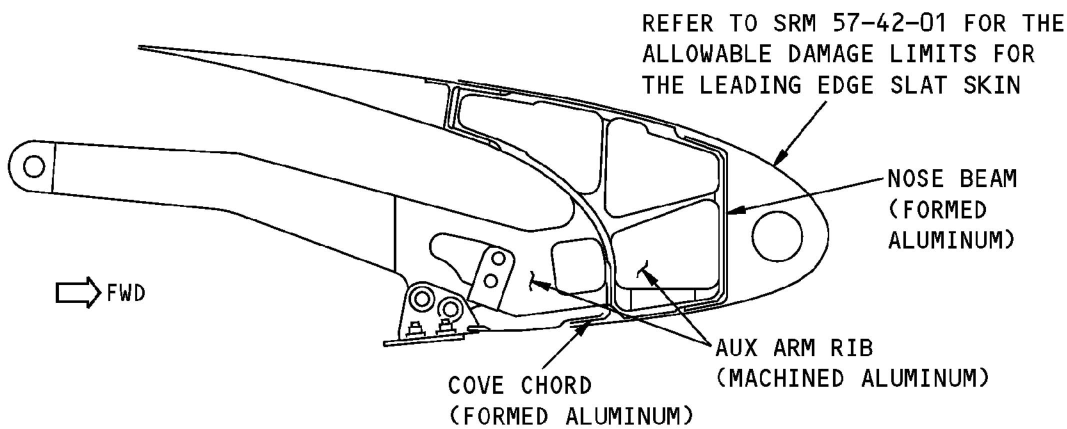



A-A

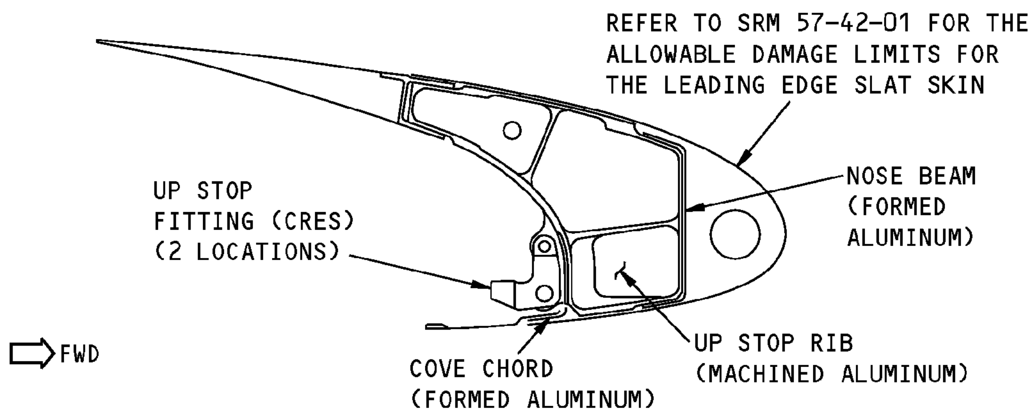



B-B

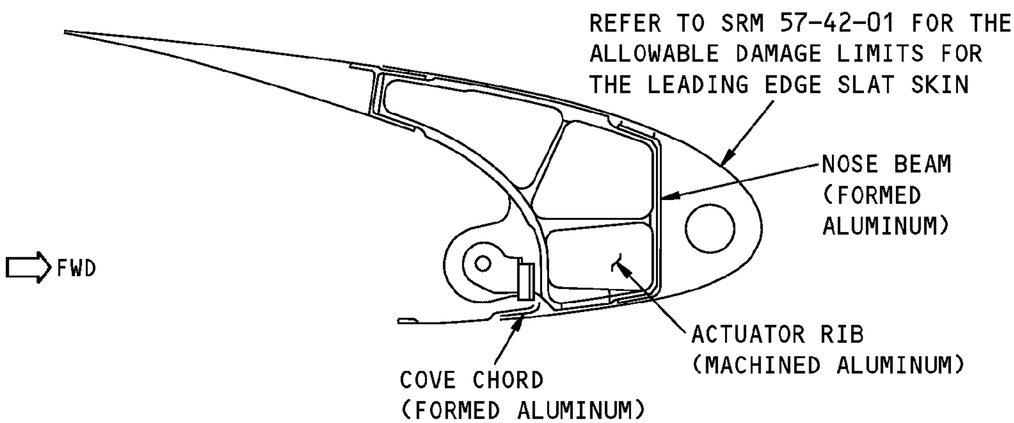
**Outboard Leading Edge Slat Structure**
**Figure 102 (Sheet 2 of 3)**




C-C




D-D




E-E

**Outboard Leading Edge Slat Structure**
**Figure 102 (Sheet 3 of 3)**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Do the steps that follow if you have damage to the aluminum, steel, or Corrosion Resistant Steel (CRES) parts:

(1) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures for the aluminum parts.

(b) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) After you remove the damage, do as follows:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(a) Flap peen or shot peen all parts except the nose beams, cove chords, end ribs, and end stop fittings.

1) Refer to 51-20-06 for the shot peen intensity and shot number.

2) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(b) Apply the finishes for the ribs.

1) Apply a chemical conversion coating to the bare surfaces. Refer to SRM 51-21-01 .

2) Apply a layer of BMS 10-79, Type III primer to the reworked surfaces. Refer to SOPM 20-44-04.

(c) Apply the finishes for the nose beams and cove chords.

1) Apply a chemical conversion coating to the bare surfaces. Refer to 51-20-01.

2) Apply a layer of BMS 10-11, Type I primer to the reworked surfaces. Refer to SOPM 20-41-02.

(d) Apply the finishes for main tracks except to the bores.

1) Apply titanium-cadmium plating to the reworked surfaces. Refer to SOPM 20-42-01.

2) Apply Tungsten Carbide with Super Detonation Gun to the roller surfaces. Refer to SOPM 20-41-01.

3) Apply a layer of BMS 10-79, Type III primer to the reworked surfaces except to the bore surfaces. Refer to SOPM 20-44-04.

4) Apply a layer of BMS 10-60, Type II enamel to the reworked surfaces except to the bore surfaces. Refer to AMM PAGEBLOCK 51-21-99/701.

(e) Apply the finishes for the fittings.

1) Apply cadmium plating to the reworked surfaces. Refer to SOPM 20-42-05.

2) Apply a layer of BMS 10-79, Type III primer to the reworked surfaces. Refer to SOPM 20-44-04.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-01 | Decoding Table For Boeing Finish Codes |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-42-01 | Low Hydrogen Embrittlement Cadmium Plating |
| SOPM 20-42-05 | Bright Cadmium Plating |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage 1 - Wing Leading Edge Slat Structure

A. Ribs (Machined Aluminum)

**NOTE**: No damage is permitted in the surfaces of the rib lug bore. You are permitted to drill the bore:

- To a maximum oversize diameter of 0.06 inch more than the initial bore design limits, and
- If there is no damage on the edge of the lug.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and G .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Cove Chords and Nose Beams (Formed Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and H .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , G , and H.

(3) Dents are not permitted.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter
2) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage
3) Filled with a 2117-T3 or 2117-T4 protruding head rivet
   a) Install the rivet without sealant.

C. Main Tracks (Forged Steel)

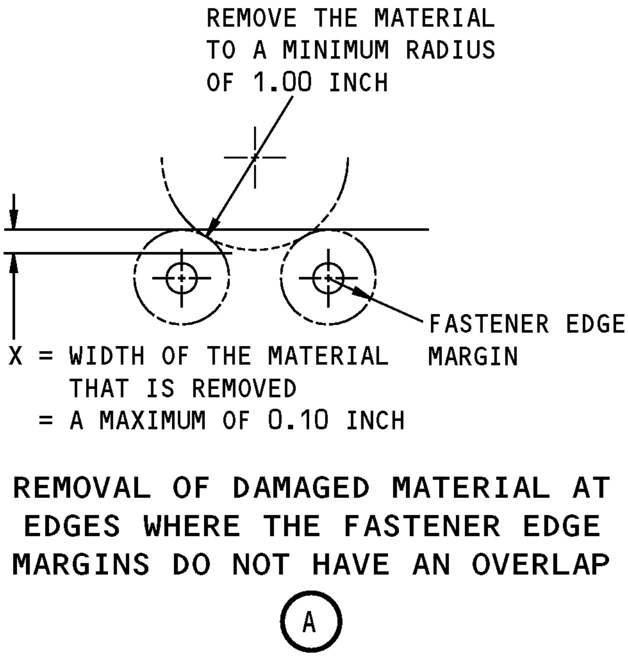
**NOTE**: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore:

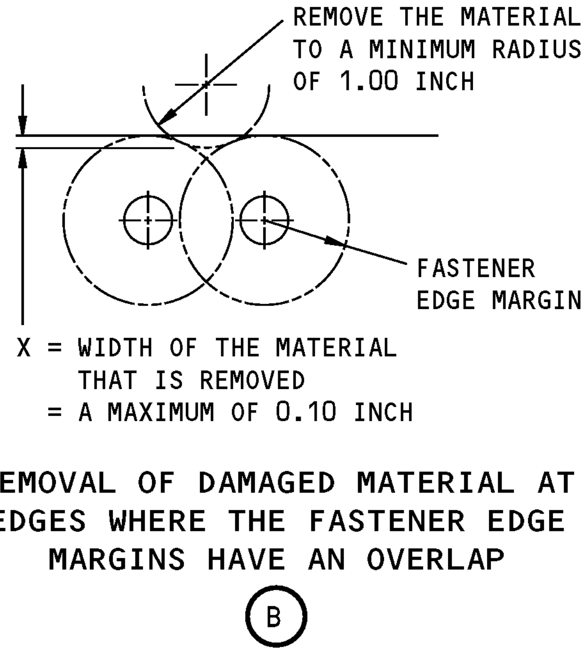
- To a maximum oversize diameter of 0.06 inch more than the initial bore design limits as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail I , and
- If there is no damage on the edge of the lug.

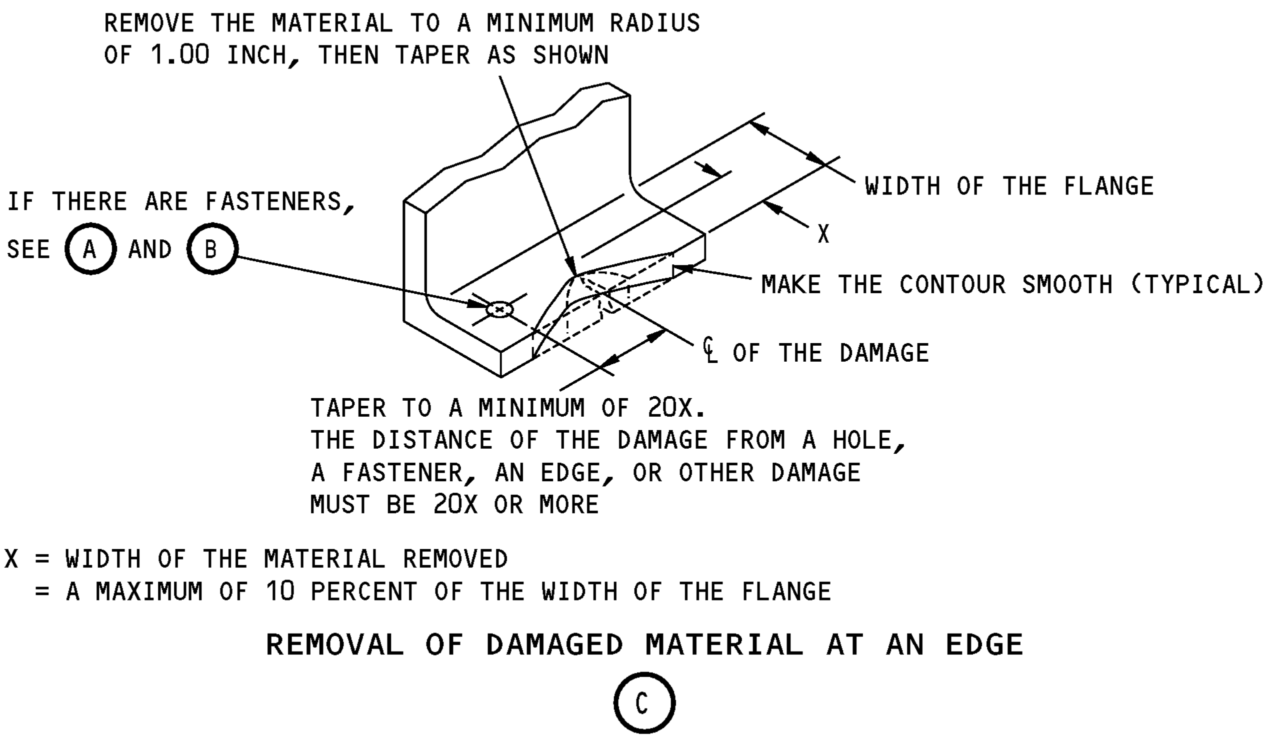
(1) Cracks are not permitted.
(2) Nicks, Gouges, Scratches, and Corrosion are not permitted.
(3) Dents are not permitted.
(4) Holes and Punctures are not permitted.
(5) Erosion is permitted if the damage is not more than the Super Detonation Gun thickness.

D. Fittings (CRES)

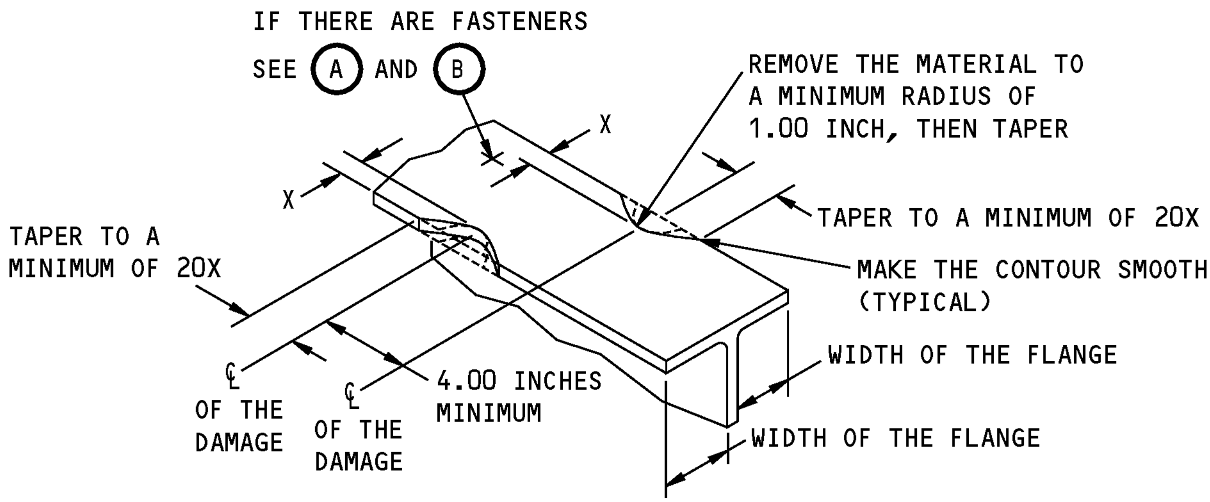
(1) Cracks are not permitted.
(2) Nicks, Gouges, Scratches, and Corrosion:
   (a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , E , and J .
(3) Dents are not permitted.
(4) Holes and Punctures are not permitted.



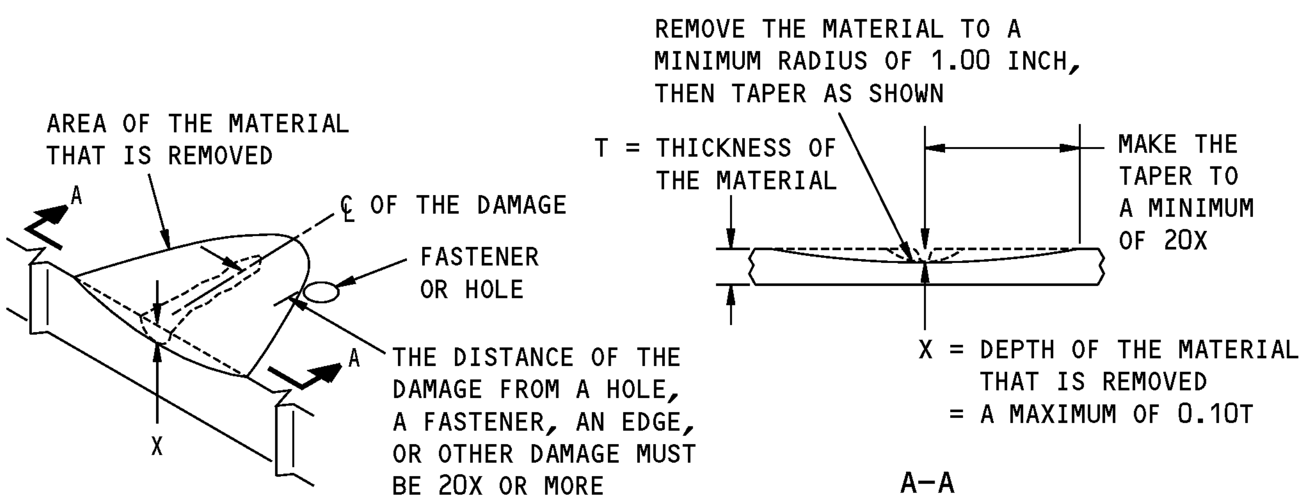

**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 7)**




X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

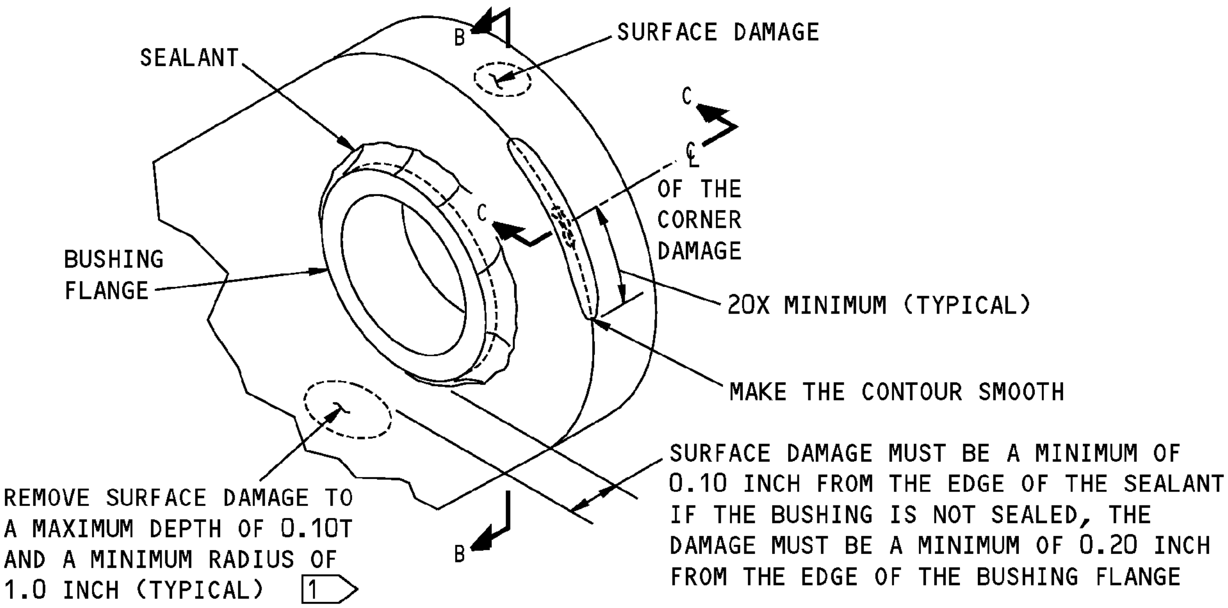
D




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

E

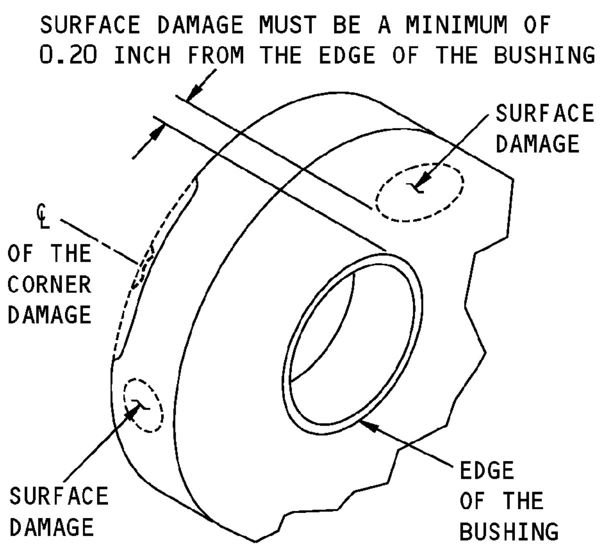
**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 7)**



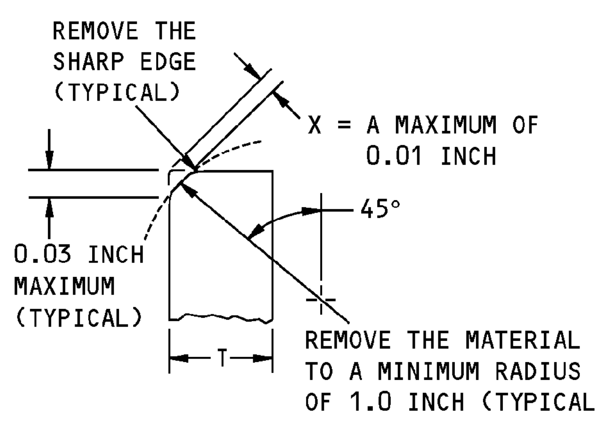

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

F




B-B

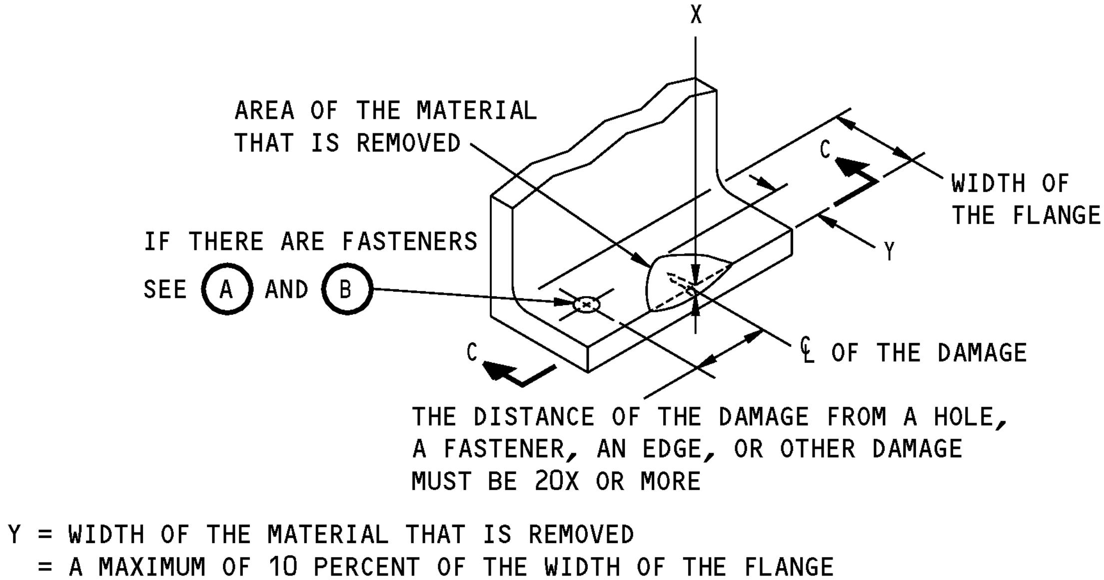



C-C 1

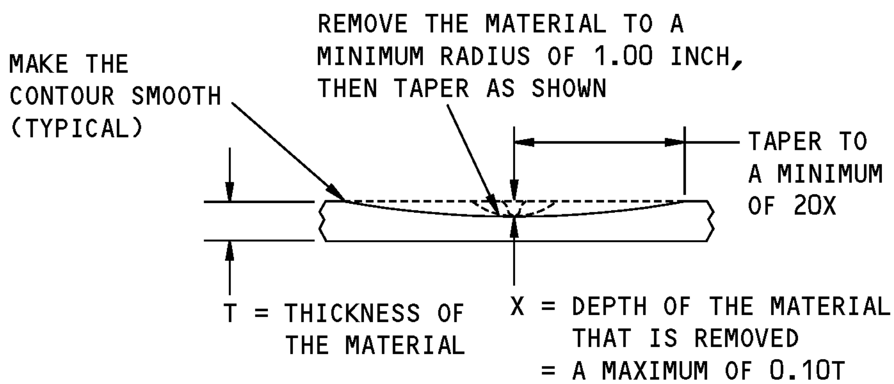
NOTES

1 THESE ALLOWABLE DAMAGE LIMITS ARE APPLICABLE ONLY IF THE BORE OF THE LUG HAS NOT BEEN OVERSIZED. IF THE BORE IS OVERSIZED, THEN ONLY SURFACE DAMAGE IS PERMITTED AS SHOWN.

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 7)**

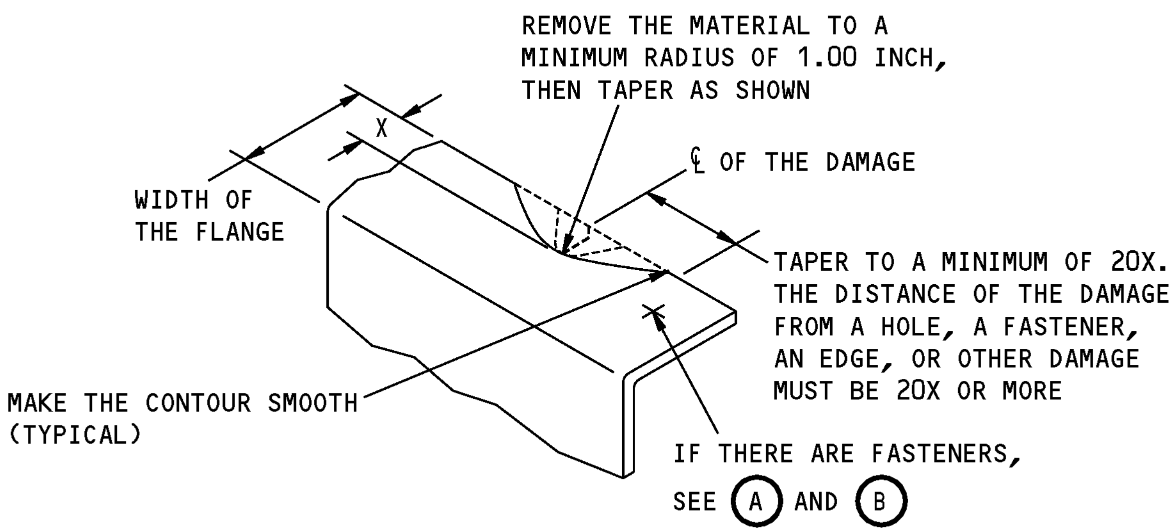



REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE
G




C-C

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 7)**

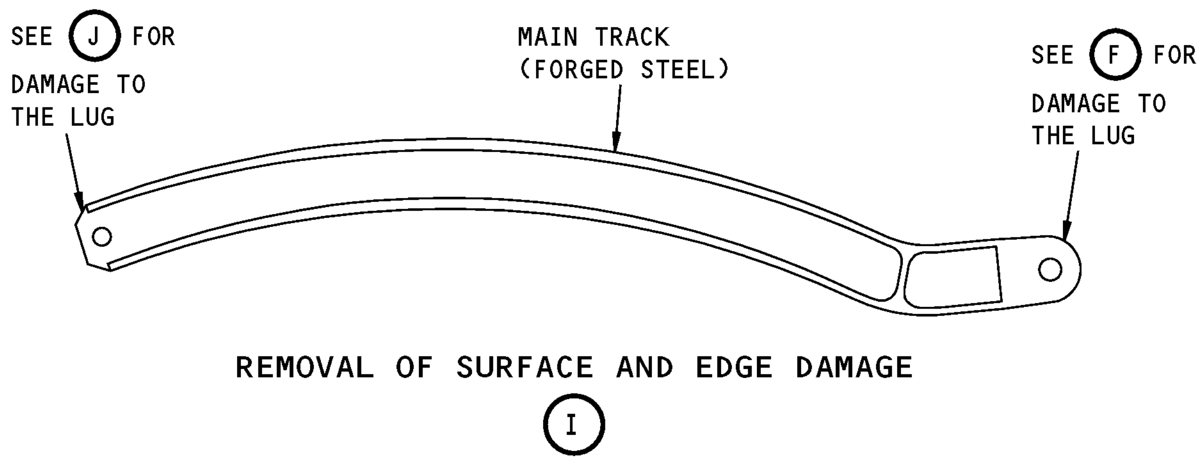



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

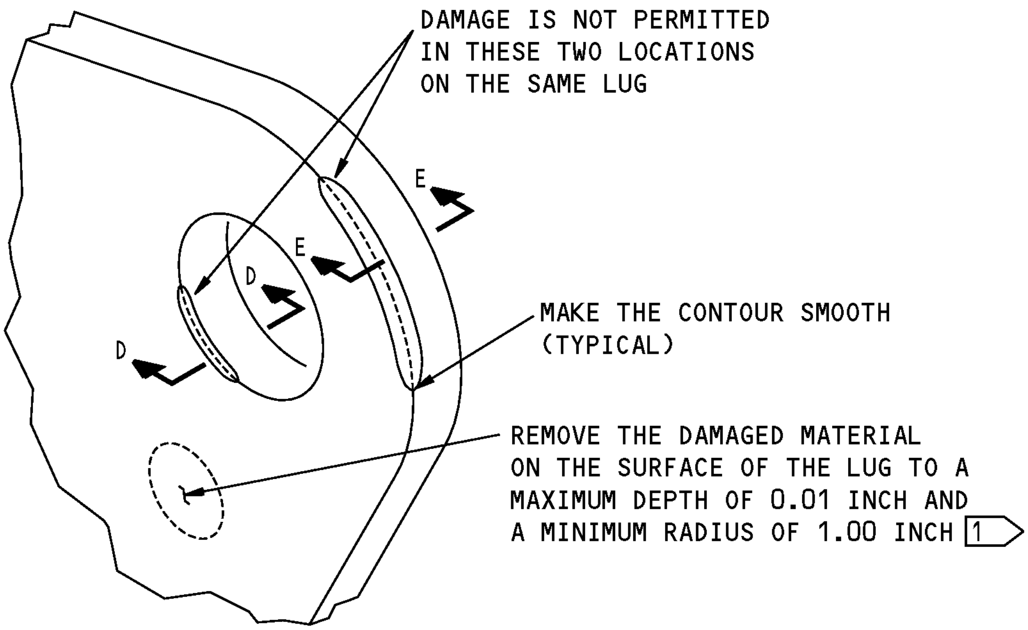
REMOVAL OF DAMAGED MATERIAL ON AN EDGE

H

**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 7)**

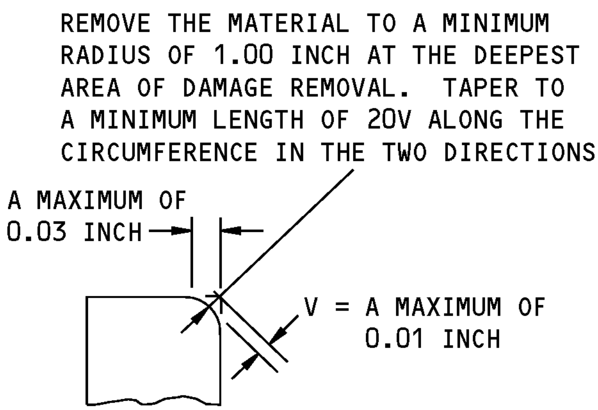



Allowable Damage Limits
Figure 103 (Sheet 6 of 7)

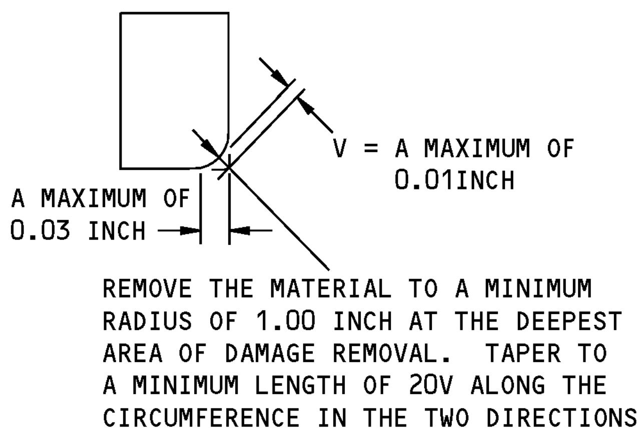



REMOVAL OF DAMAGED MATERIAL ON A LUG WITHOUT BUSHINGS

J




D-D 1
(ROTATED)




E-E 1

**Allowable Damage Limits**
**Figure 103 (Sheet 7 of 7)**

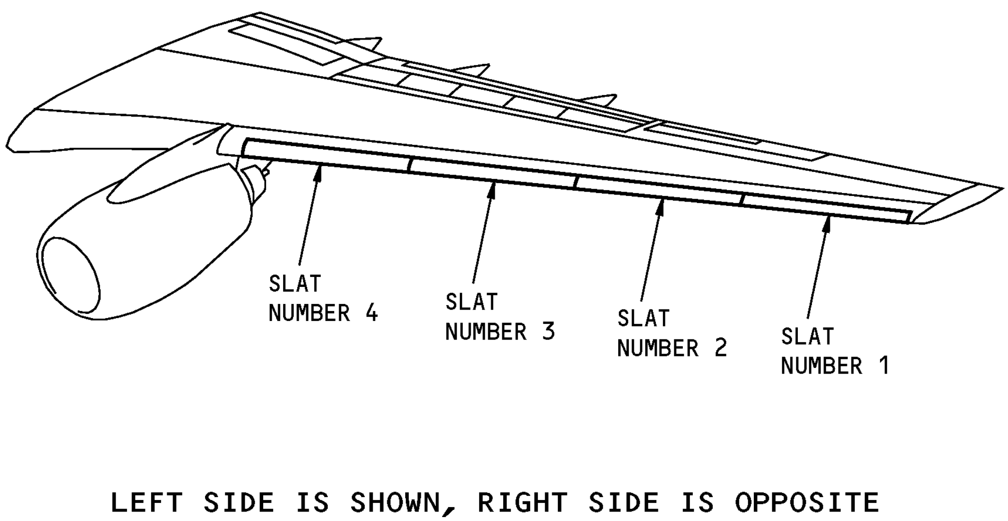
## REPAIR 1 - WING LEADING EDGE SLAT STRUCTURE AT A NOSE BEAM

### 1. Applicability

A. Repair 1 is applicable to damage:

(1) To the structure of the wing leading edge slat shown in Wing Leading Edge Slat Skin Locations, Figure 201/REPAIR 1.

(2) That is at the nose beam.




**Wing Leading Edge Slat Skin Locations**
**Figure 201**

### 2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Blind rivets that are loose, damaged, or that have fallen out must be replaced.

C. Repair 1 can only be used if you install the blind rivets flush against the internal structure.

**3. References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

**4. Repair Instructions**

A. Cut and remove the damaged part of the leading edge slat nose beam. Refer to 51-10-02 for the damage removal procedures.

(1) Make the cut in the shape of a rectangle.

(a) Refer to Wing Leading Edge Slat Structure Repair at a Nose Beam, Figure 202/REPAIR 1 for a typical cut out.

B. Make the repair parts as shown in Wing Leading Edge Slat Structure Repair at a Nose Beam, Figure 202/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair parts.

**Table 201**:

| REPAIR MATERIALS | | | |
|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **QUANTITY** | **MATERIAL** |
| [1] | Doubler | 1 | Use 7075-T6 clad sheet that is 0.080 inch thick |
| [2] | Angle Filler (For the upper flange only) | 1 | Use 7075-T6 clad sheet that is 0.071 inch thick |
| [3] | Angle Filler (For the lower flange only) | 1 | Use 7075-T6 clad sheet that is 0.071 inch thick |
| [4] | Spacer Filler (For the upper flange only) | 6 | Use 7075-T6 clad sheet that is 0.125 inch thick |
| [5] | Repair Angle (For the upper flange only) | 1 | Use 7075-T6 clad sheet that is 0.080 inch thick |

C. Assemble the repair parts.

D. Drill the necessary fastener holes.

E. Disassemble the repair parts.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and to the bare surfaces of the initial structure.

G. Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

H. Apply one layer of BMS 10-11, Type I primer to:

(1) The surfaces of the initial structure

(2) The mating surfaces of the repair parts.

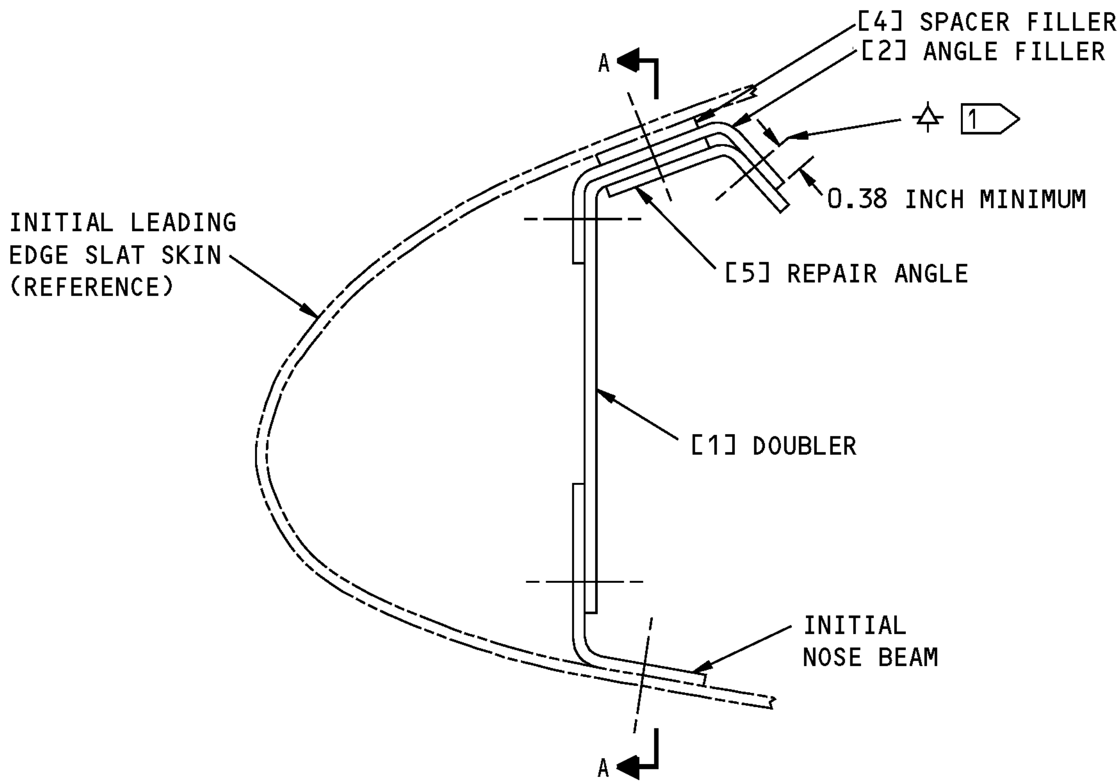
I. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

J. Install the fasteners.

(1) Install the aluminum fastener without sealant.

(2) Install the fasteners that are not made of aluminum with BMS 5-95 sealant. Refer to 51-20-05.

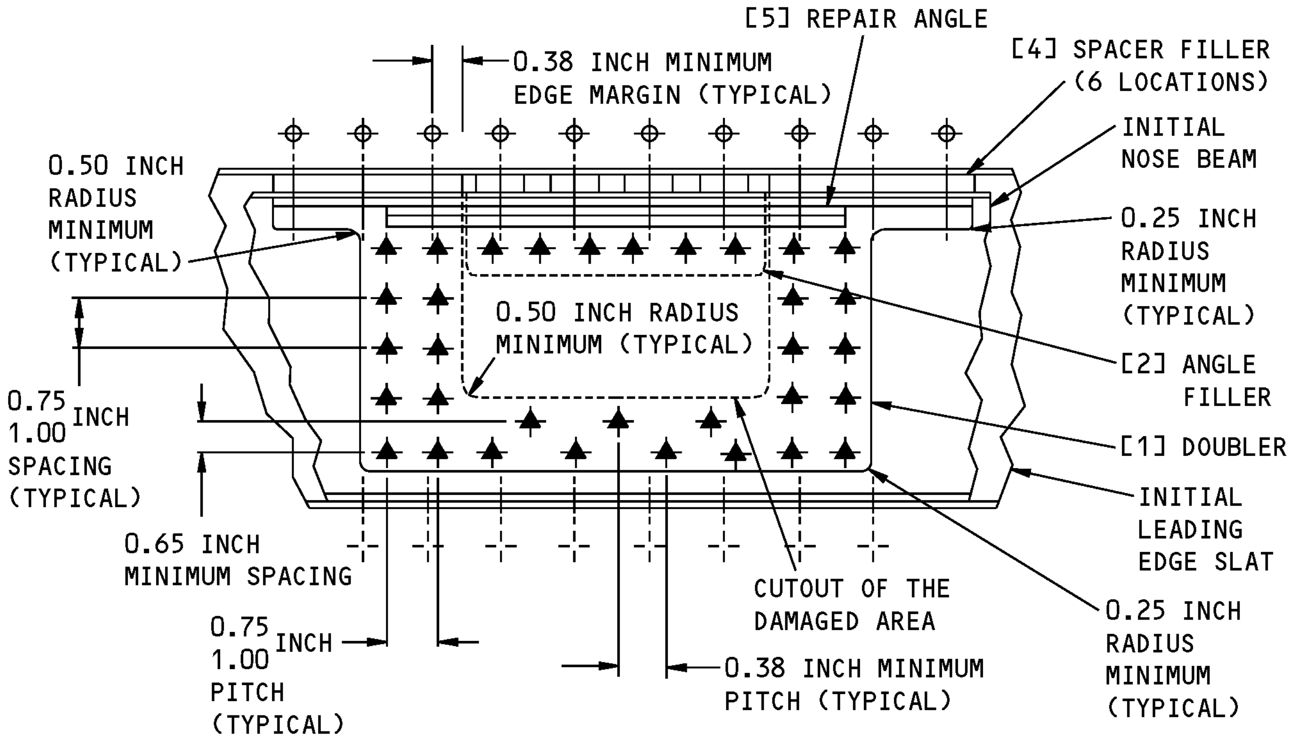
K. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.




A TYPICAL SECTION IS SHOWN

1 INSTALL THE FASTENERS ALONG THE SAME SPAN LOCATION AS THE INITIAL UPPER SKIN FASTENERS. DO NOT INSTALL THE FASTENERS INTO THE REPAIR ANGLE BEND RADIUS.

**Wing Leading Edge Slat Structure Repair at a Nose Beam**
**Figure 202 (Sheet 1 of 4)**




UPPER FLANGE DAMAGE
A-A

FASTENER SYMBOLS

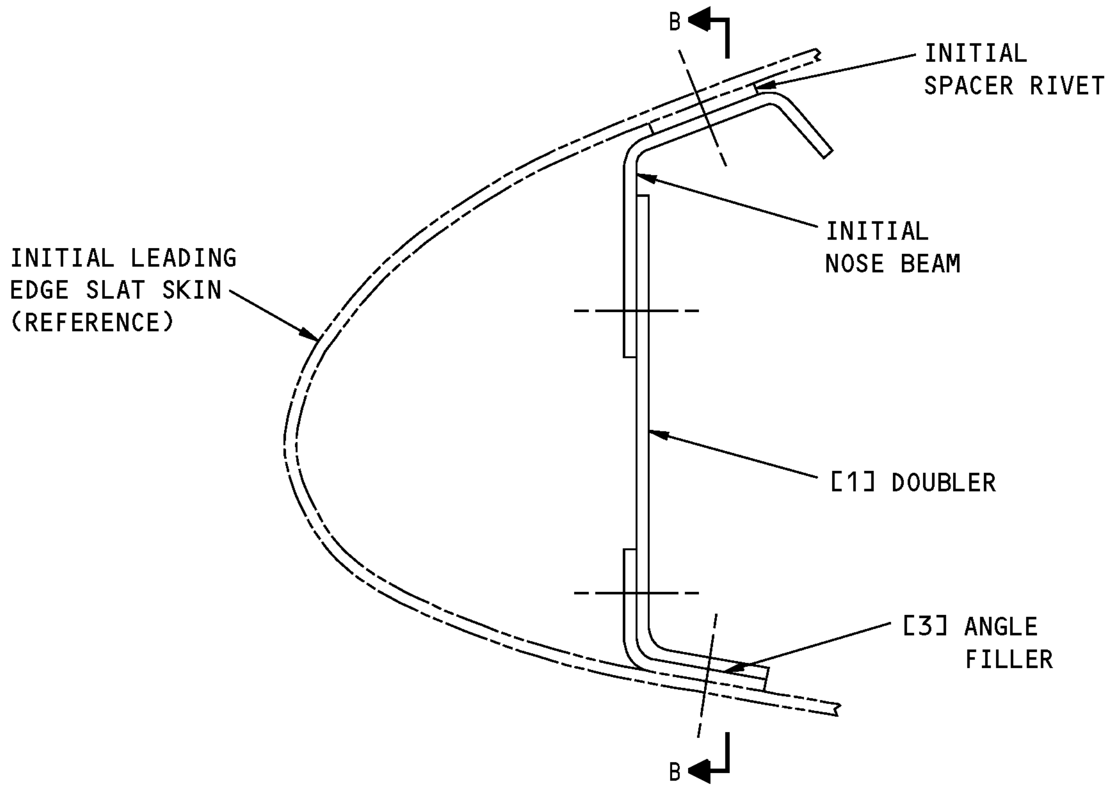
⌖ INITIAL FASTENER LOCATION. INSTALL A BACB30NW6K HEX-DRIVE BOLT WITH A BACC30M COLLAR.

▲ REPAIR FASTENER LOCATION. INSTALL A NAS1738E6 BLIND RIVET.

△ REPAIR FASTENER LICATION. INSTALL A BACR15BB6D RIVET.

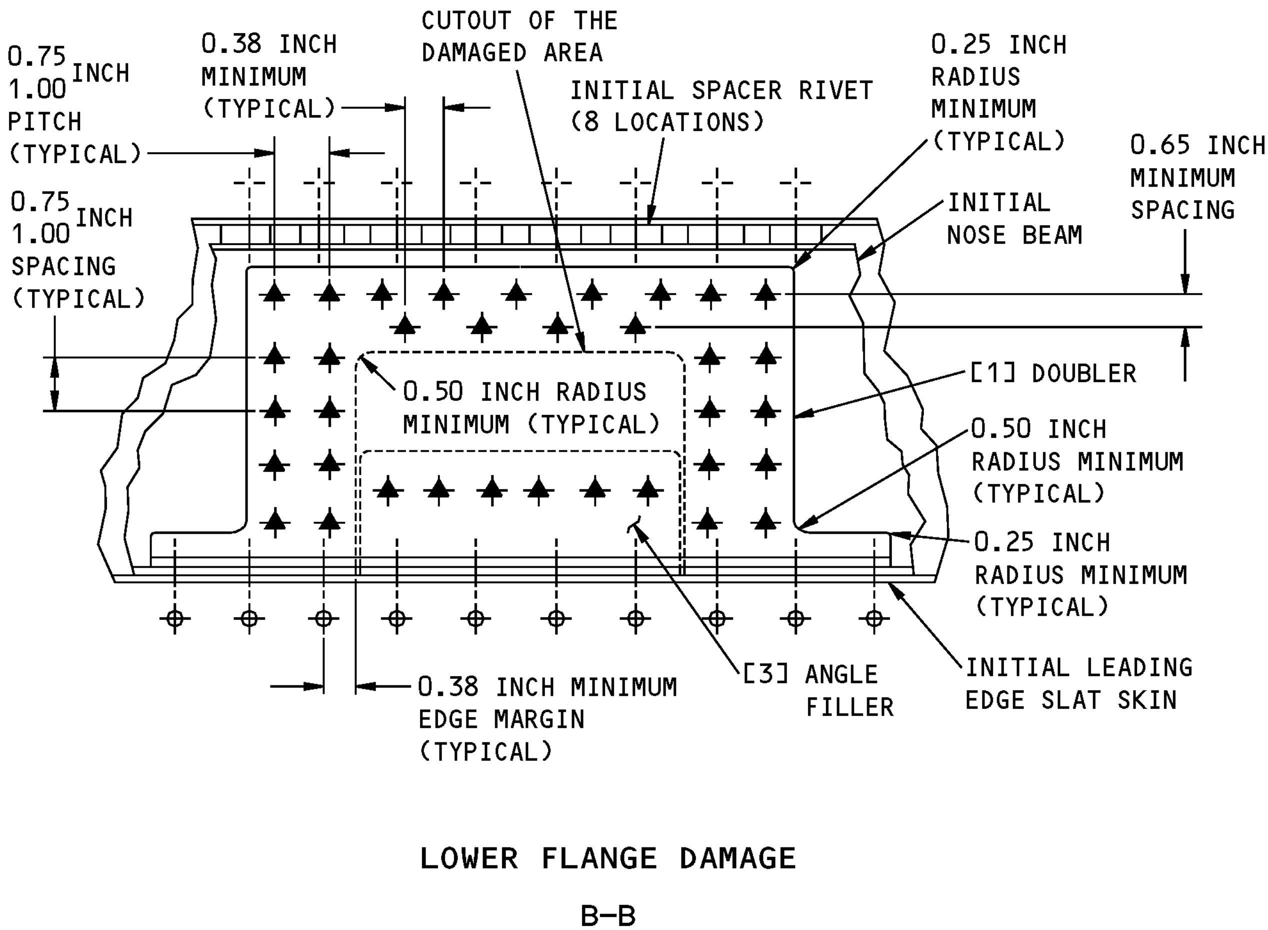
-¦- REFERENCE FASTENER

**Wing Leading Edge Slat Structure Repair at a Nose Beam**
**Figure 202 (Sheet 2 of 4)**



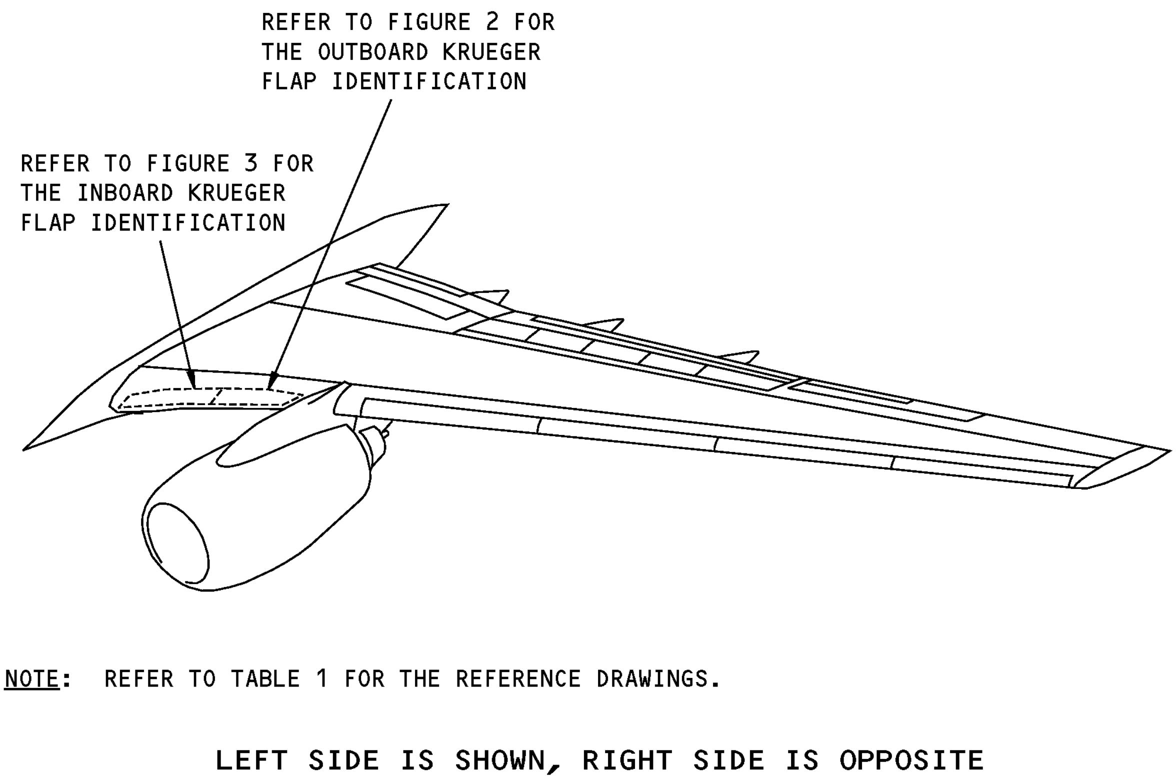

A TYPICAL SECTION IS SHOWN

**Wing Leading Edge Slat Structure Repair at a Nose Beam**
**Figure 202 (Sheet 3 of 4)**




**Wing Leading Edge Slat Structure Repair at a Nose Beam**
**Figure 202 (Sheet 4 of 4)**

## IDENTIFICATION 1 - INBOARD AND OUTBOARD WING LEADING EDGE - KRUEGER FLAP STRUCTURE

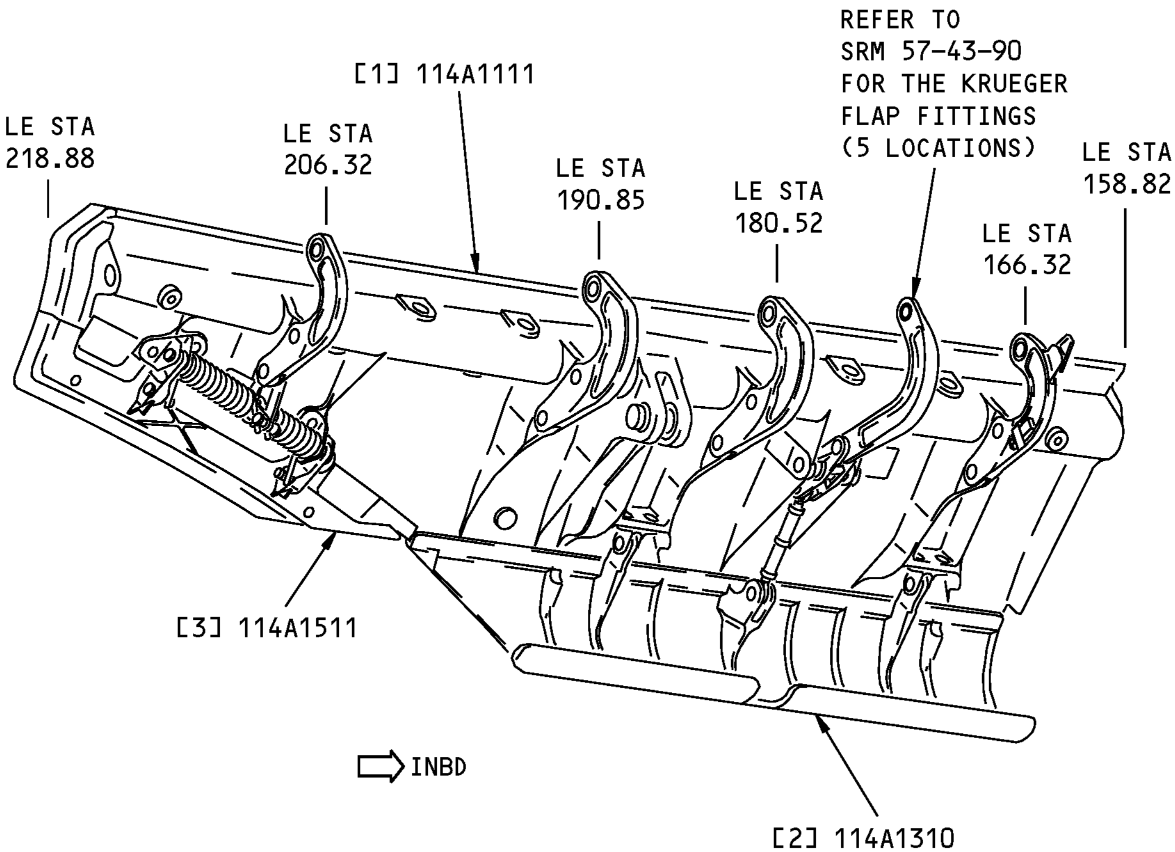



**Inboard Wing Leading Edge Krueger Flap Structure Locations**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 114A0010 | Section 14 Final Assembly |
| 114A1001 | Krueger Flap Installation - Inboard Wing Leading Edge |
| 114A1110 | Krueger Flap and Bullnose Assembly - Number 1 and Number 4 |
| 114A1220 | Krueger Flap and Bullnose Assembly - Number 2 and Number 3 |
| 114A1221 | Flap Assembly - Inboard Krueger Flap |
| 114A1320 | Bullnose Assembly - Middle, Krueger Flap Number 2 |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 114A1321 | Bullnose Assembly Inboard Krueger Flap Number 2 |
| 114A1401 | Rod Assembly - Krueger Flap, Inboard Wing Leading Edge |
| 114A1410 | Link Assembly - Outboard Bullnose, Krueger Flap, L. E. |
| 114A1411 | Link Assembly Bullnose, Outboard and Middle |
| 114A1413 | Bell Crank Assembly - Bullnose |
| 114A1420 | Link Assembly - Middle Bullnose Krueger Flap L. E., Wing |
| 114A1430 | Link Assembly - Inboard Bullnose Krueger Flap L. E., Wing |
| 114A1510 | Krueger Seal Assembly - Inboard Fixed Leading Edge |
| 114A4001 | Slat Rigging Installation - Outboard Leading Edge Wing |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

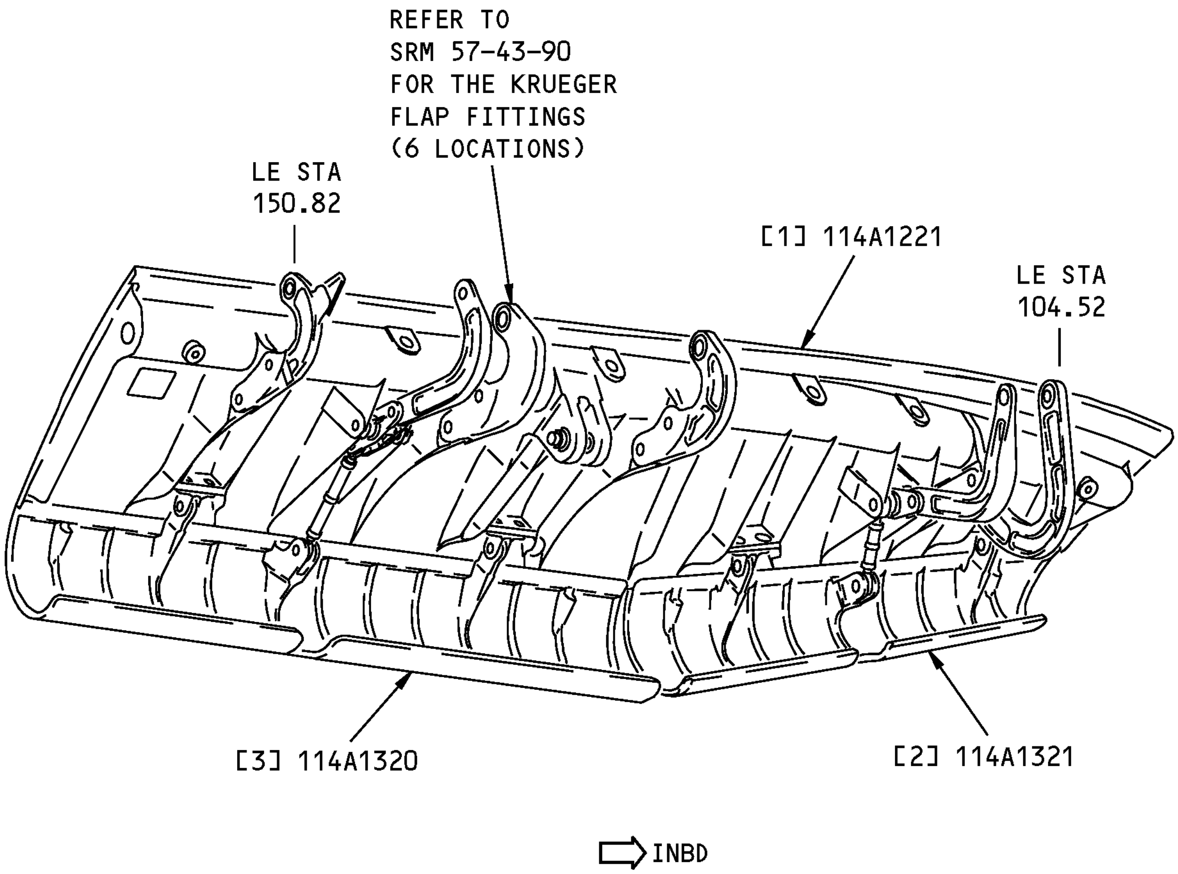
NUMBER 1 KRUEGER FLAP IS SHOWN,
NUMBER 4 KRUEGER FLAP IS OPPOSITE

**Outboard Krueger Flap Structure Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Krueger Flaps - Number 1 and Number 4 | | A357-T6 aluminum casting, as given in MIL-A-21180 | |
| [2] | Bullnose, Outboard Flap | | A357-T6 aluminum casting, as given in MIL-A-21180 | |
| [3] | Krueger - Seal | | A357-T6 aluminum casting, as given in MIL-A-21180 | |

*[1] Note: T = Pre-manufactured thicknesses in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

NUMBER 2 KRUEGER FLAP IS SHOWN,
NUMBER 3 KRUEGER FLAP IS OPPOSITE

**Inboard Krueger Flap Structure Identification**
**Figure 3**

**Table 3**:

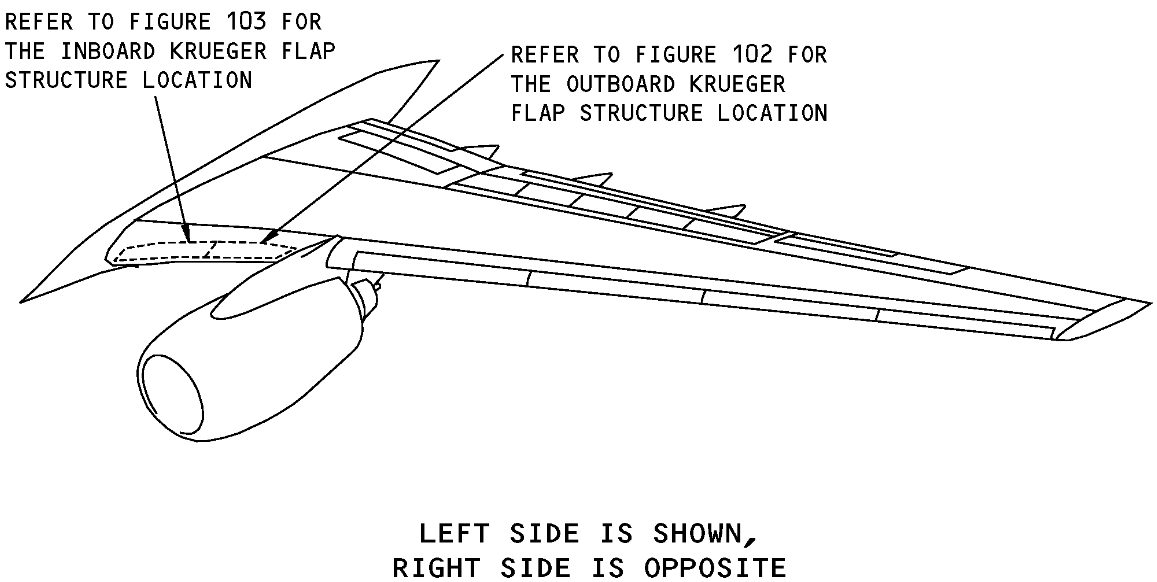
| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Krueger Flaps - Number 2 and Number 3 | | A357-T6 aluminum casting, as given in MIL-A-21180 | |
| [2] | Bullnose, Inboard Flap - Number 2 | | A357-T6 aluminum casting, as given in MIL-A-21180 | |
| [3] | Bullnose, Middle Flap - Number 2 | | A357-T6 aluminum casting, as given in MIL-A-21180 | |

*[1] Note: T = Pre-manufactured thicknesses in inches (millimeters).

## ALLOWABLE DAMAGE 1 - WING LEADING EDGE KRUEGER FLAP STRUCTURE

### 1. Applicability

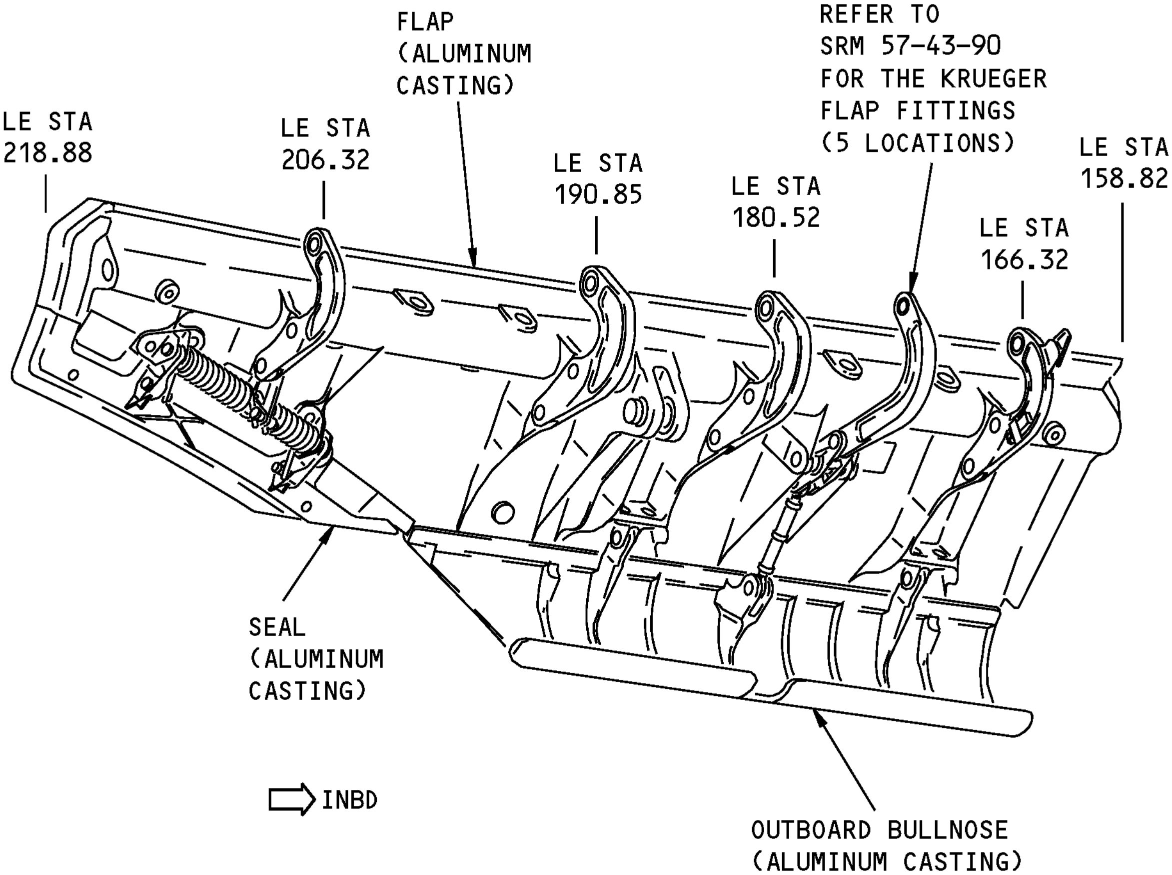
A. This subject gives the allowable damage limits for the wing leading edge Krueger flap structure shown in Inboard Wing Leading Edge Krueger Flap Structure Locations, Figure 101/ALLOWABLE DAMAGE 1.




**Inboard Wing Leading Edge Krueger Flap Structure Locations**
**Figure 101**

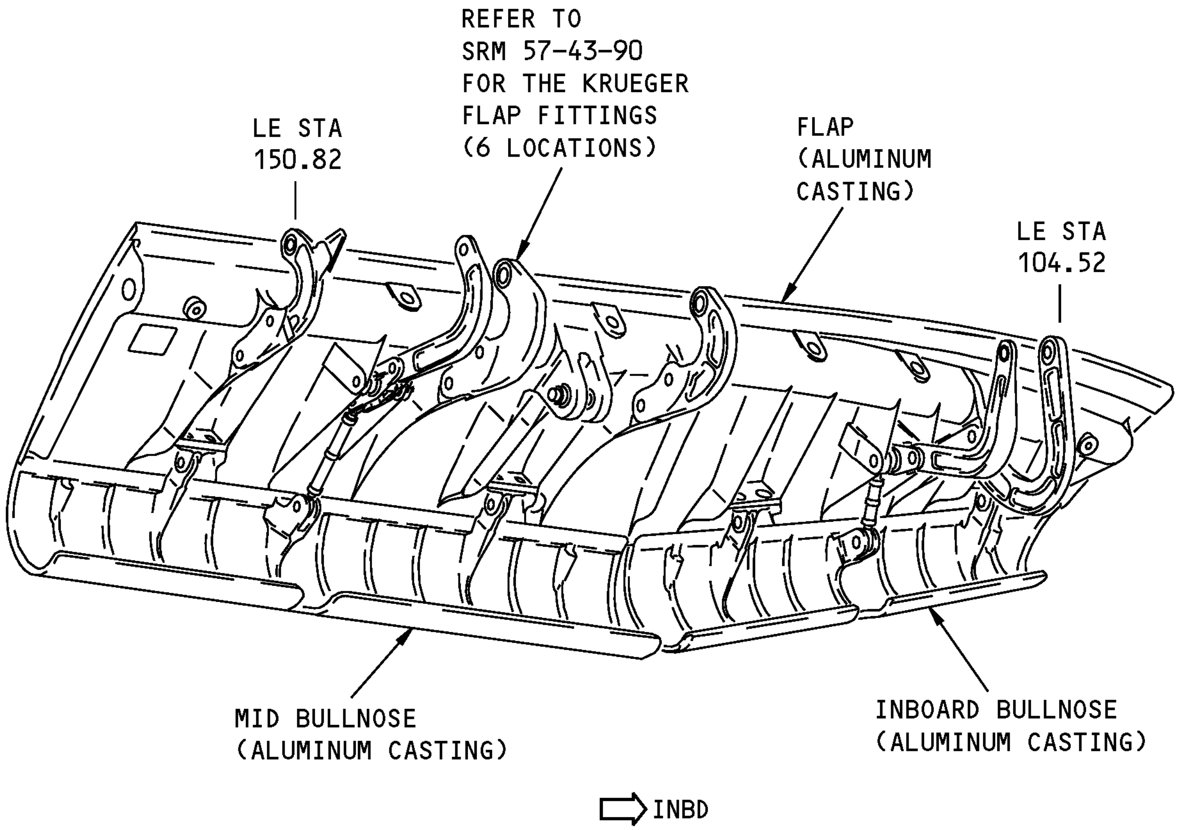
### 2. General

A. Do the steps that follow if you have damage to the aluminum parts:
   (1) Do a penetrant inspection of the damaged area. Refer to SOPM 20-20-02.
   (2) Remove the damage as necessary.
      (a) Refer to 51-10-02 for the investigation and cleanup procedures.
      (b) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.
      (c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.
   (3) After you remove the damage, do the procedures that follow:
      (a) Apply a chemical conversion coating to the bare surfaces of the aluminum parts. Refer to 51-20-01.
      (b) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.
      (c) Make sure the aerodynamic smoothness is satisfactory or there will be a loss in economic performance. Refer to 51-10-01.




THE NUMBER 1 KRUEGER FLAP IS SHOWN,
THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

**Outboard Krueger Flap Structure Location**
**Figure 102**




THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

**Inboard Krueger Flap Structure Location**
**Figure 103**

## 3. References

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Seals, Flaps, and Bullnoses

(1) Cracks:

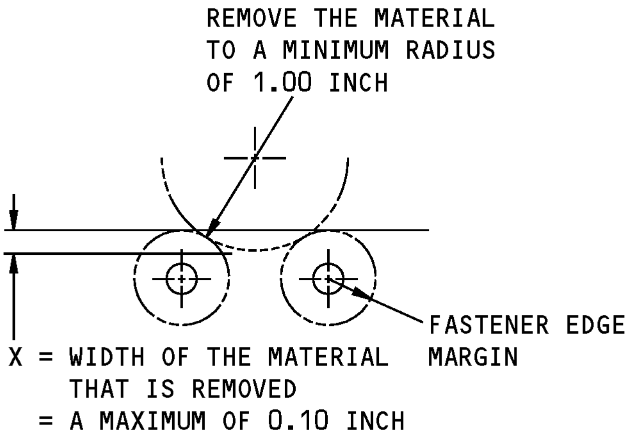
(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and F.

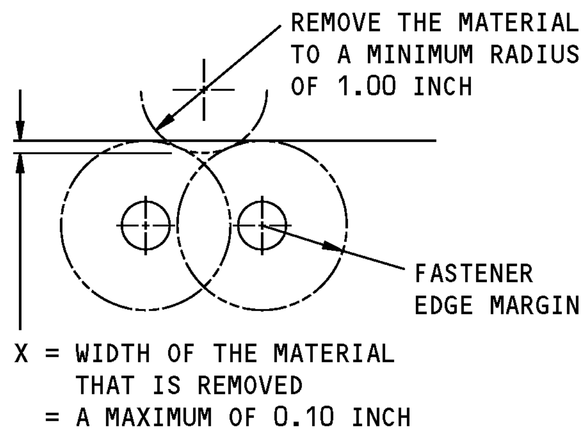
(3) Dents are permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail G .

(4) Holes and Punctures are not permitted.



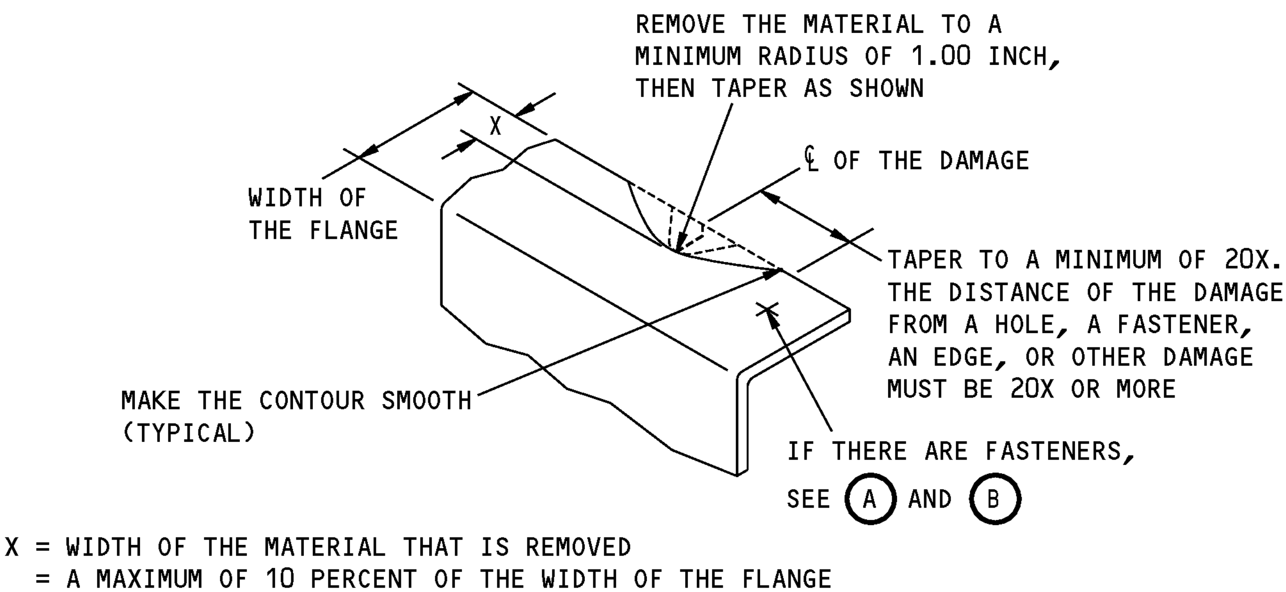

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

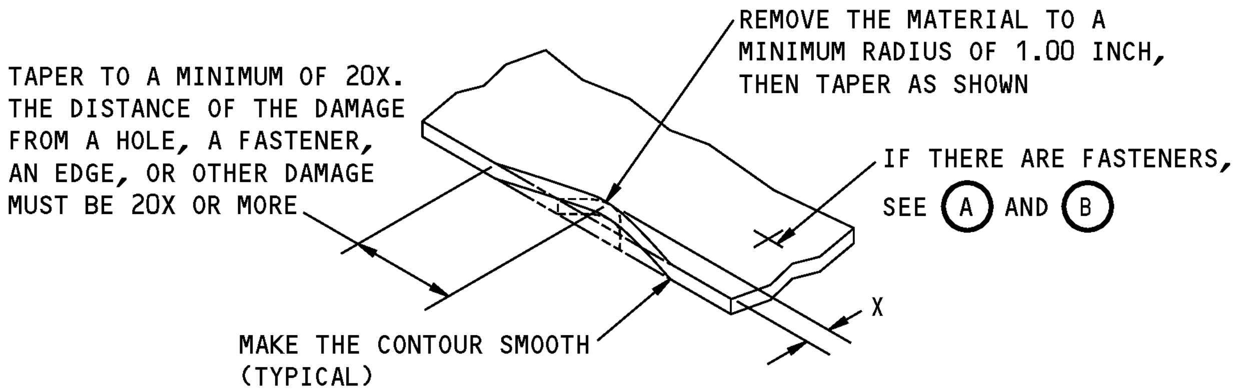
B




REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

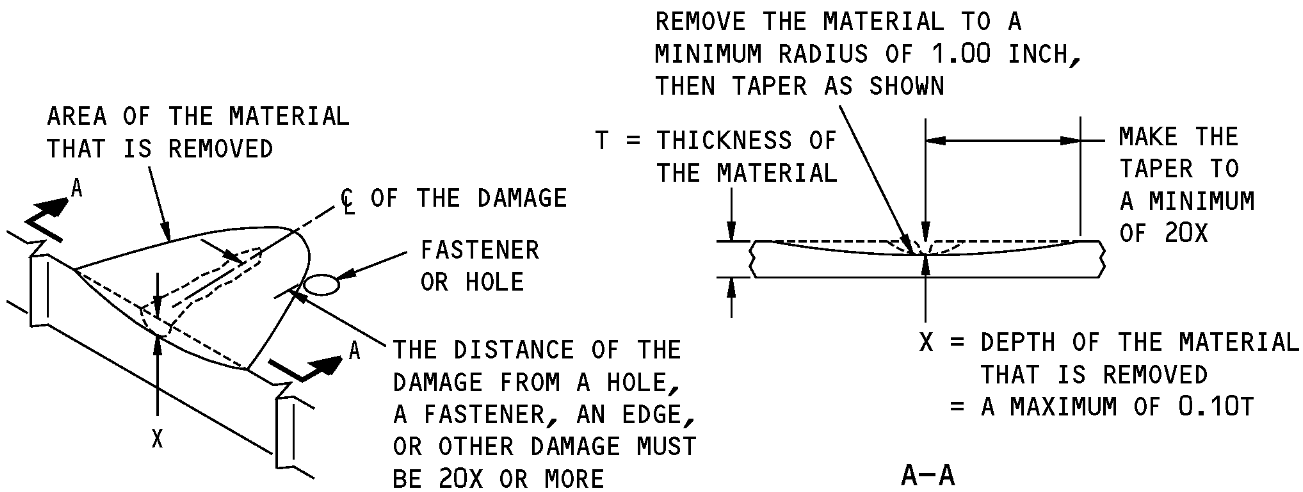
**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**




X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

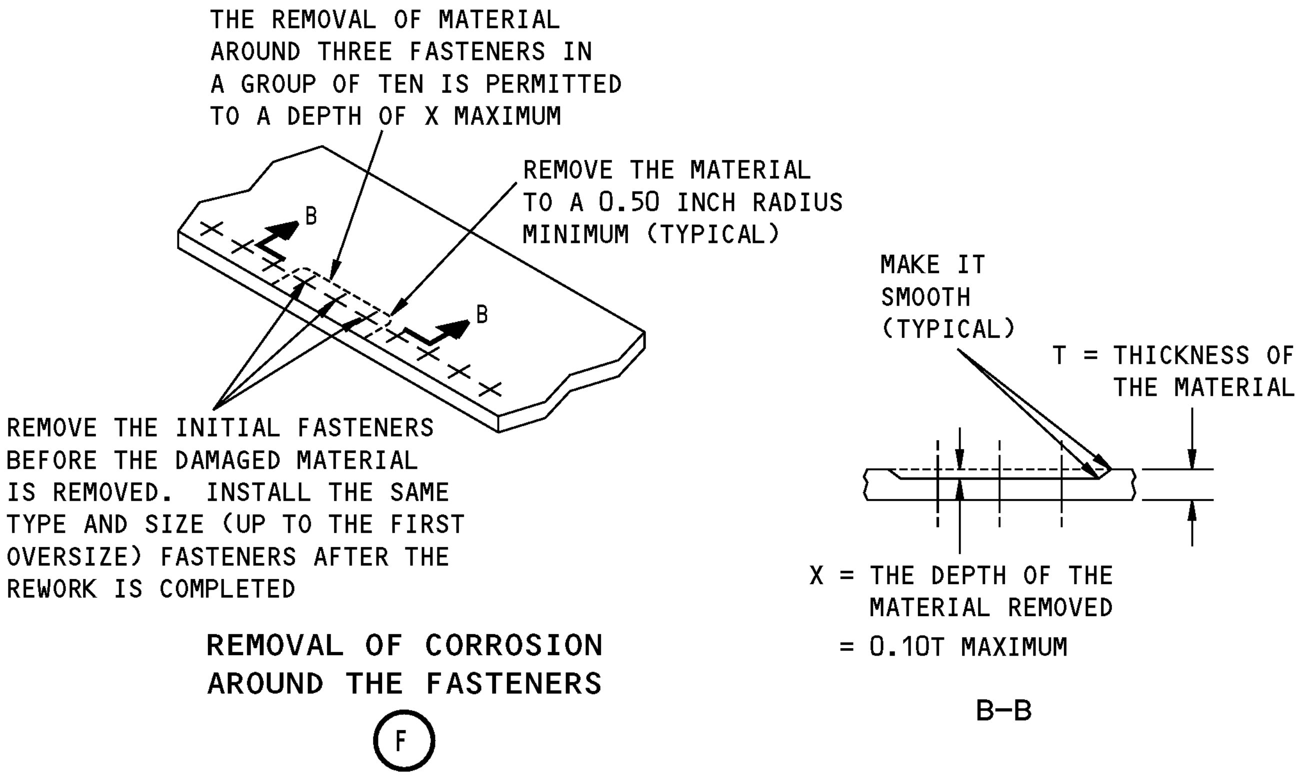
D

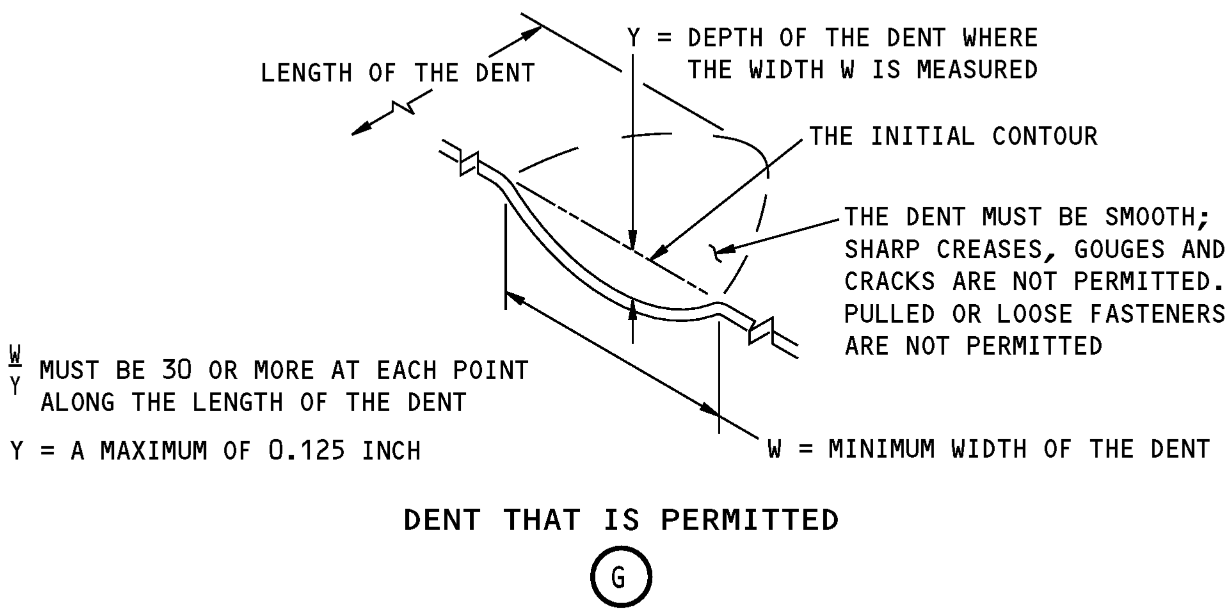



REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

E

**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**




**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**

## REPAIR 1 - INBOARD WING LEADING EDGE KRUEGER FLAP STRUCTURE - EXTERNAL REPAIR BETWEEN THE RIBS

### 1. Applicability

A. Repair 1 is applicable to damage:

(1) To the structure of the wing leading edge Krueger flap

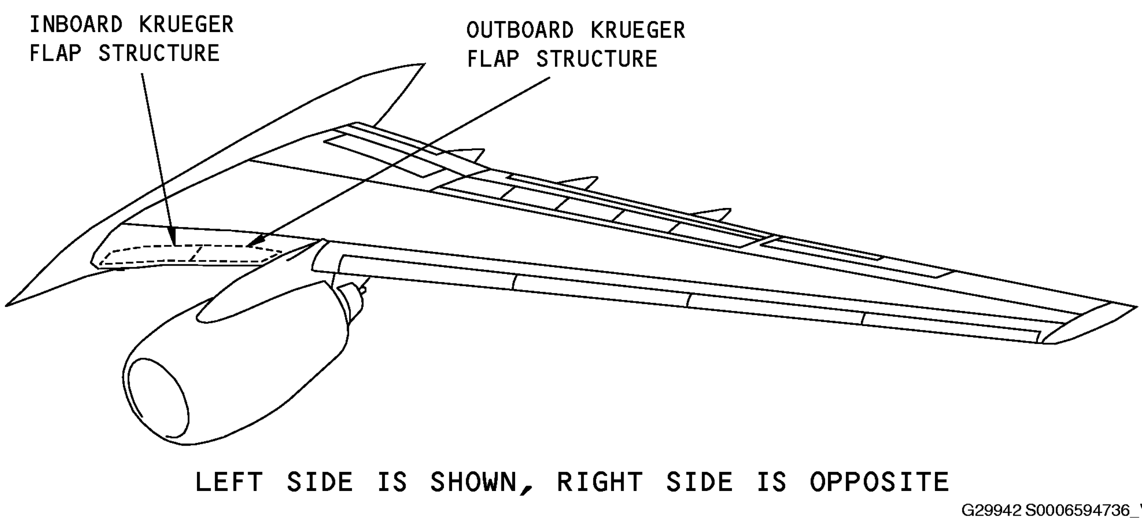
(2) That is between the ribs as shown in Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1

(3) Where it is not possible to make a satisfactory weld repair to the damaged area.

### 2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the economic performance of the airplane. Refer to 51-10-01.




**Wing Leading Edge Krueger Flap Structure Locations**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |

(Continued)

| Reference | Title |
|---|---|
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-50-11 | Application of Aerodynamic Smoothing Sealant |

4. **Repair Instructions**

A. Cut and remove the damaged part of the leading edge flap structure. Refer to 51-10-02 for the damage removal procedures.

(1) Make the cut in the shape of a rectangle with the sides perpendicular to the rib.

(a) Refer to Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1 for a typical cutout.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

(a) Refer to Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1 for the shape of the cutout.

B. Make the repair parts as shown in Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1. Refer to Table 201/REPAIR 1 for the materials.

(1) Make sure there is a minimum of two rows of repair fasteners around the edge of the cutout.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Internal Doubler | 1 | Use clad 2024-T3 sheet that is 0.050 inch thick |
| [2] | External Doubler | 1 | Use clad 2024-T3 sheet that is 0.071 inch thick |
| [3] | Filler | 1 | Use clad 2024-T3 sheet that is the same thickness as the initial leading edge flap structure |

C. Assemble the repair parts and drill the fastener holes.

(1) Do not countersink the fastener holes more than 70 percent of the material thickness.

(a) This will prevent a knife-edge condition of the external structure.

D. Disassemble the repair parts.

E. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial structure.

F. Apply one layer of BMS 10-79, Type III primer to:

(1) The surfaces of the initial structure

(2) The mating surfaces of the repair parts.

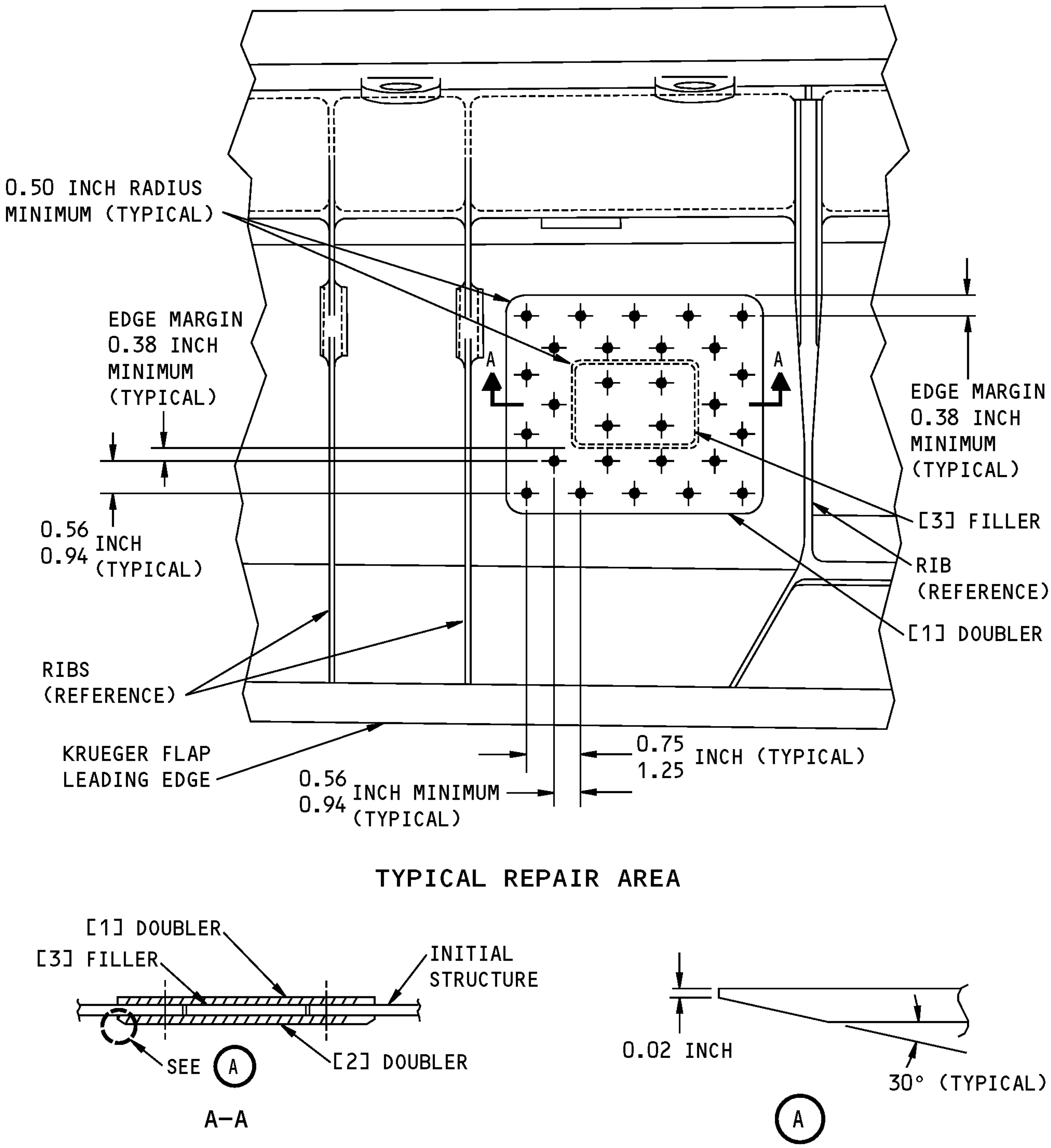
G. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

(1) Fill the space between the part [3] filler and the initial structure with BMS 5-95 sealant.

H. Install the fasteners by the squeeze rivet method and without sealant.

**NOTE**: It is optional to install the fasteners by the hand drive method. Make sure to visually inspect for cracks in the initial structure.

I. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.




FASTENER SYMBOLS

-◆- REPAIR FASTENER LOCATION. INSTALL A BACR15BA6AD RIVET.

**Krueger Flap Structure Wing Leading Edge Repair Between Ribs**
**Figure 202**

## REPAIR 2 - WING LEADING EDGE KRUEGER FLAP STRUCTURE - EXTERNAL REPAIR AT RIBS

### 1. Applicability

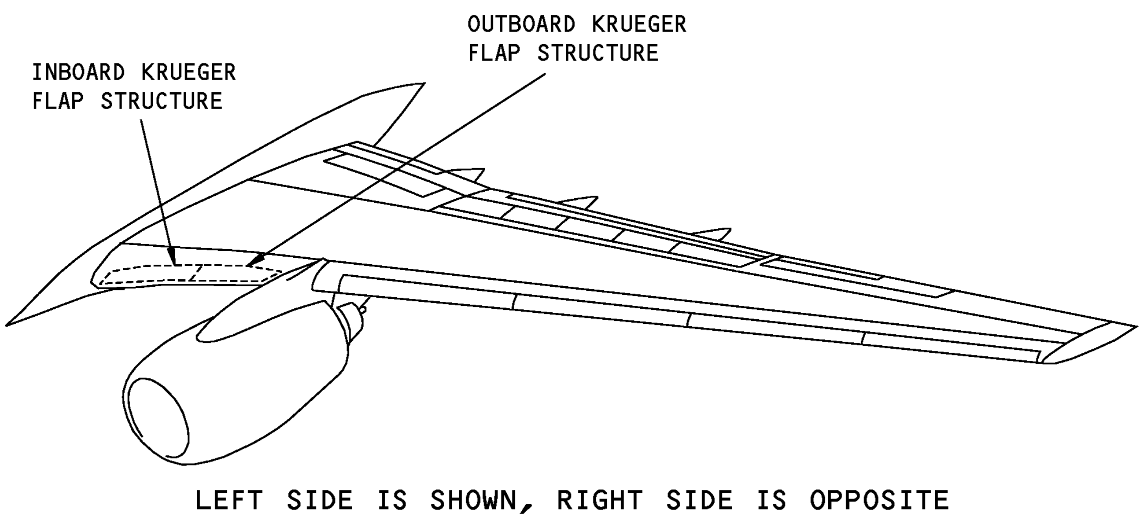
A. Repair 2 is applicable to damage:

(1) To the structure of the wing leading edge Krueger flap

(2) That is at the ribs as shown in Krueger Flap Structure Wing Leading Edge Repair at a Rib, Figure 202/REPAIR 2.

### 2. General

A. Repair 2 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.




**Wing Leading Edge Krueger Flap Structure Locations**
**Figure 201**

### 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |

(Continued)

| Reference | Title |
|---|---|
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-50-11 | Application of Aerodynamic Smoothing Sealant |

**4. Repair Instructions**

A. Cut and remove the damaged part of the leading edge flap structure. Refer to 51-10-02 for the removal of damage procedures.

(1) If possible, remove the tapered part of the rib.

(a) Install a tapered filler if the tapered rib can not be removed.

(2) Make the cut in the shape of a rectangle.

(a) Refer to Krueger Flap Structure Wing Leading Edge Repair at a Rib, Figure 202/REPAIR 2 for a typical cutout.

B. Make the repair parts as shown in Krueger Flap Structure Wing Leading Edge Repair at a Rib, Figure 202/REPAIR 2. Refer to Table 201/REPAIR 2 for the repair parts.

(1) Make sure you can put a minimum of two rows of repair fasteners around the edge of the cutout.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Angle | 2 | Use clad 2024-T4 sheet that is 0.063 inch thick |
| [2] | Filler | As necessary | Use clad 2024-T3 clad. Use a maximum thickness of 0.03 inch to fill the space between the installed part [1] angle and the initial rib. Taper the filler as necessary |
| [3] | Filler | 1 | Use 2024-T3 clad sheet that is the same thickness as the initial rib |
| [4] | Filler | 1 | Use 2024-T3 clad sheet that is the same thickness as the initial rib |
| [5] | Doubler | 1 | Use 2024-T3 clad sheet that is 0.071 inch thick |

C. Assemble the repair parts and drill the fastener holes.

(1) Do not countersink the fastener holes more than 70 percent of the material thickness.

(a) This will prevent a knife-edge condition of the initial structure.

D. Disassemble the repair parts.

E. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and to the bare surfaces of the initial structure.

F. Apply one layer of BMS 10-79, Type III primer to:

(1) The surfaces of the initial structure

(2) The mating surfaces of the repair parts.

G. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

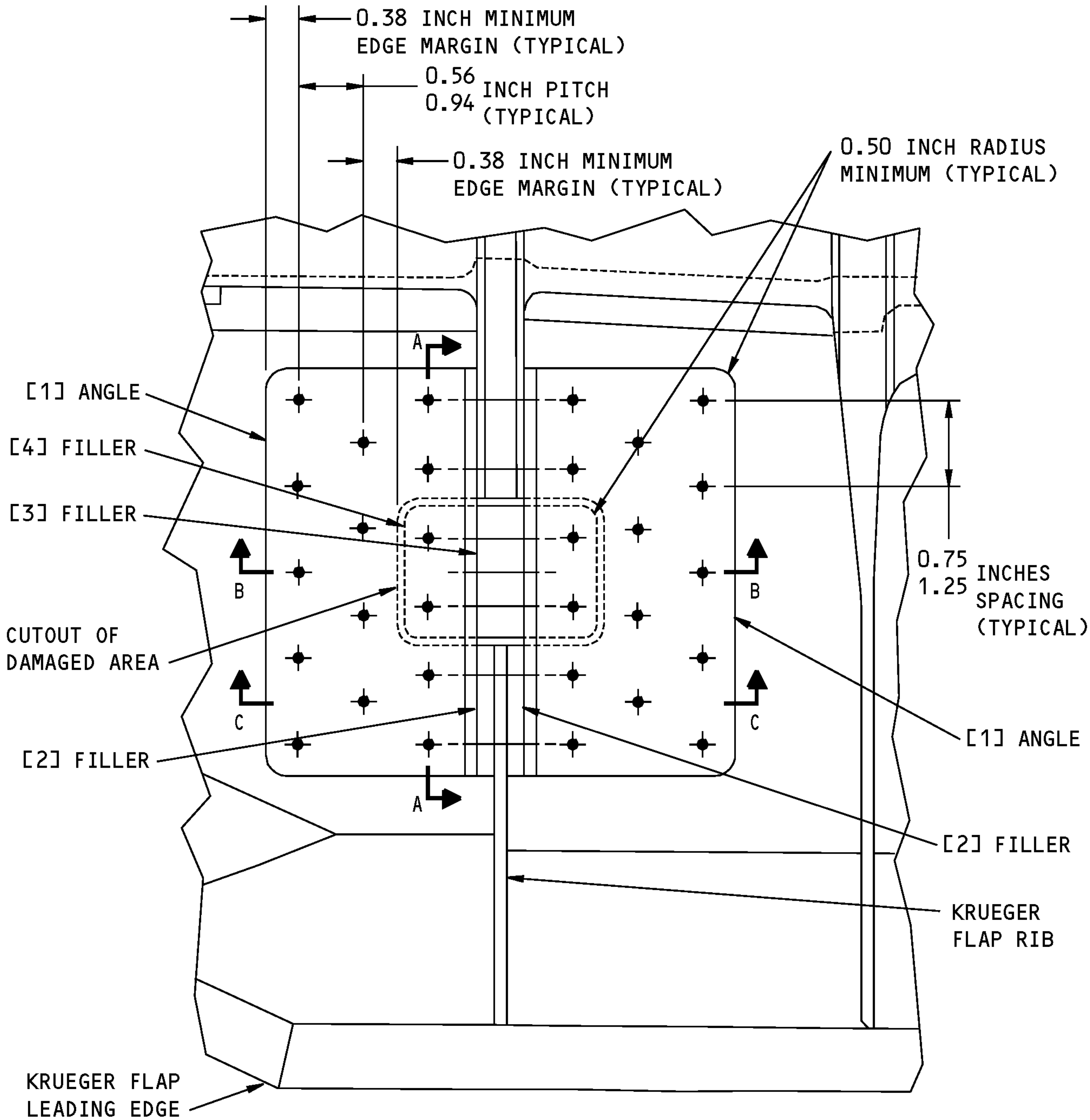
**NOTE**: As necessary, fill the space between the part [2] filler and the mating surfaces with shims.

(1) Fill the spaces between the part [2] and part [3] fillers and the initial structure with BMS 5-95 sealant.

H. Install the fasteners by the squeeze rivet method and without sealant. Refer to 51-40-02.

**NOTE**: It is optional to install the fasteners by the hand drive method. Make sure to visually inspect for any cracks to the initial structure.

I. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.



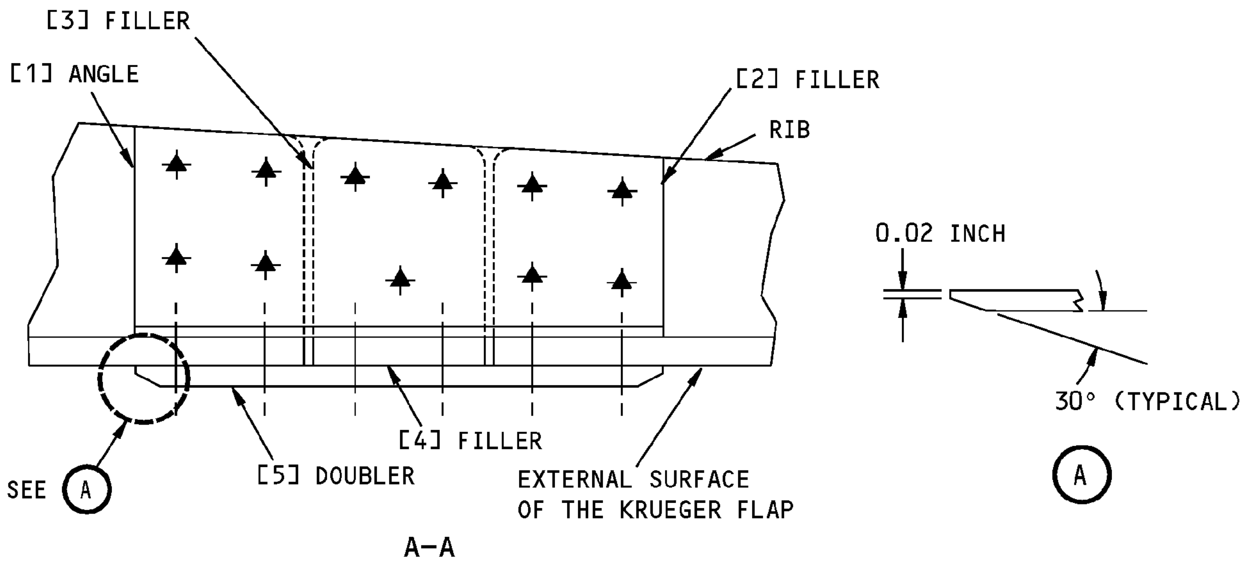

TYPICAL REPAIR AREA

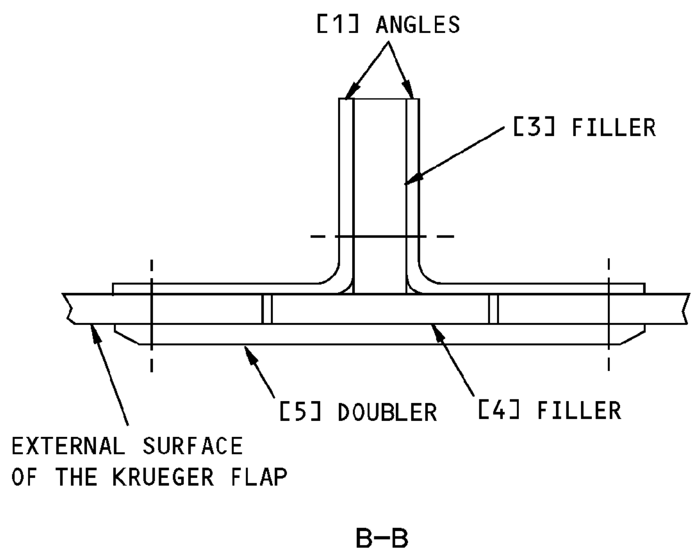
FASTENER SYMBOLS

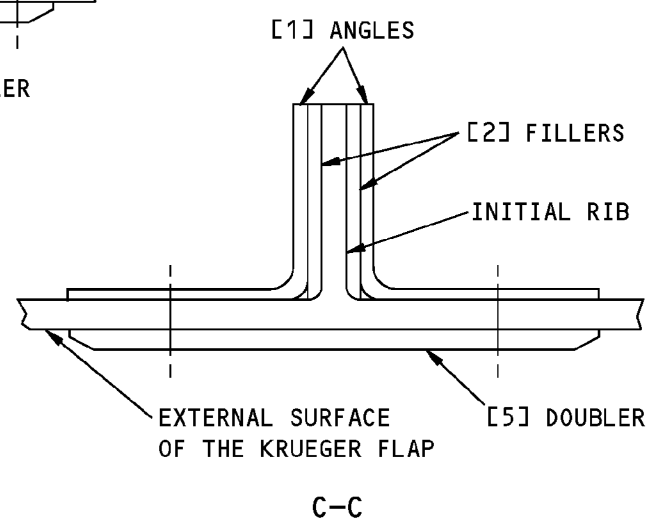
REPAIR FASTENER LOCATION. INSTALL A BACR15CE6D RIVET.

REPAIR FASTENER LOCATION. INSTALL A BACR15BB6D RIVET.

**Krueger Flap Structure Wing Leading Edge Repair at a Rib**
**Figure 202 (Sheet 1 of 2)**

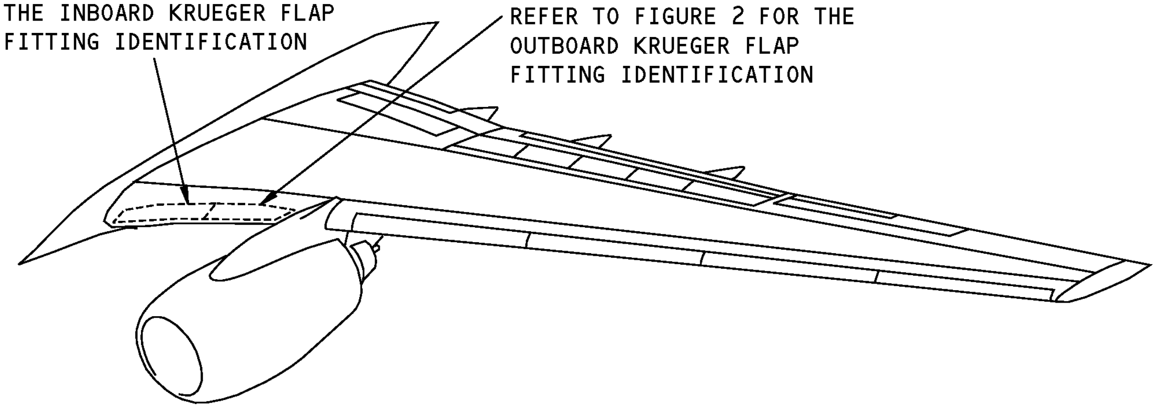



**Krueger Flap Structure Wing Leading Edge Repair at a Rib**
**Figure 202 (Sheet 2 of 2)**

## IDENTIFICATION 1 - INBOARD WING LEADING EDGE (KRUEGER) FLAP FITTINGS



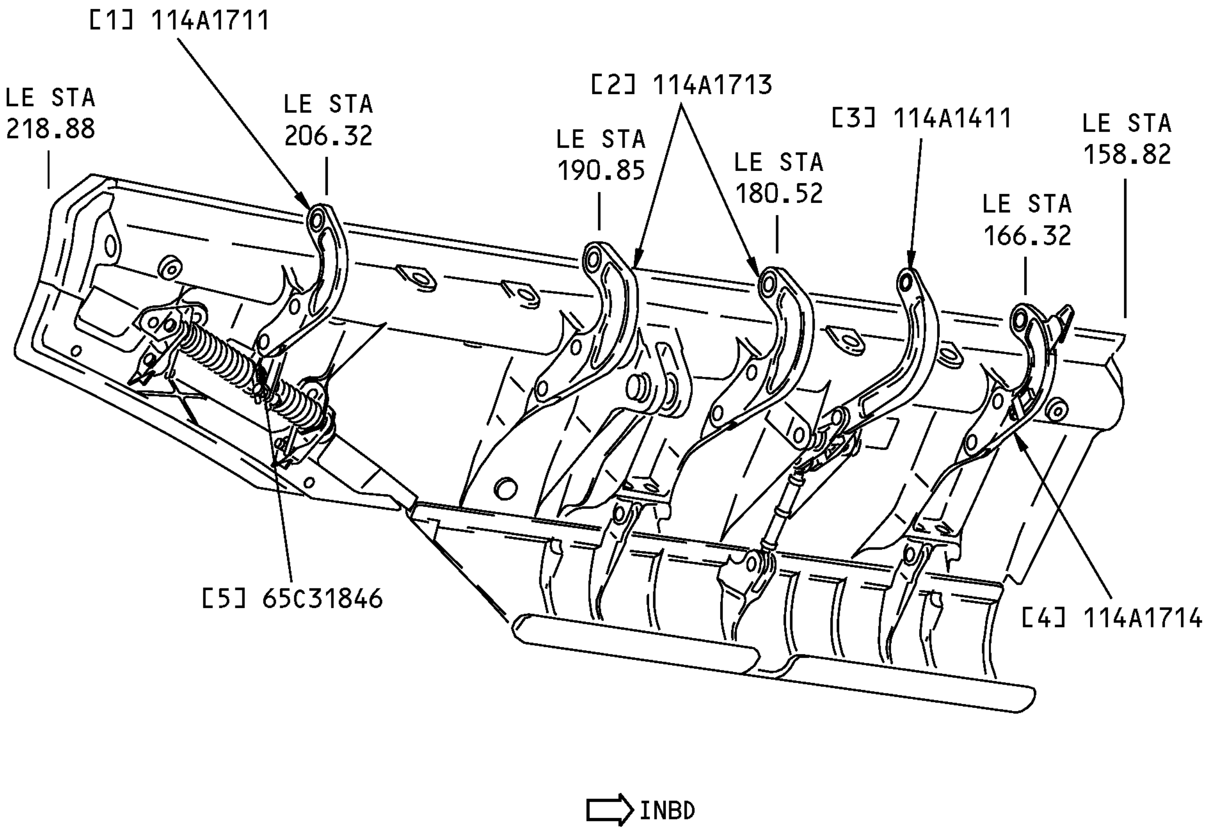

NOTE: REFER TO TABLE 1 FOR THE LIST OF REFERENCE DRAWINGS.

THE LEFT SIDE WING LEADING EDGE FLAPS ARE SHOWN,
THE RIGHT SIDE WING LEADING EDGE FLAPS ARE OPPOSITE

**Inboard Wing Leading Edge (Krueger) Flap Fitting Locations**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 114A1110 | Krueger Flap and Bullnose Assembly - Numbers 1 and 4 |
| 114A1220 | Krueger Flap and Bullnose Assembly - Numbers 2 and 3 |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

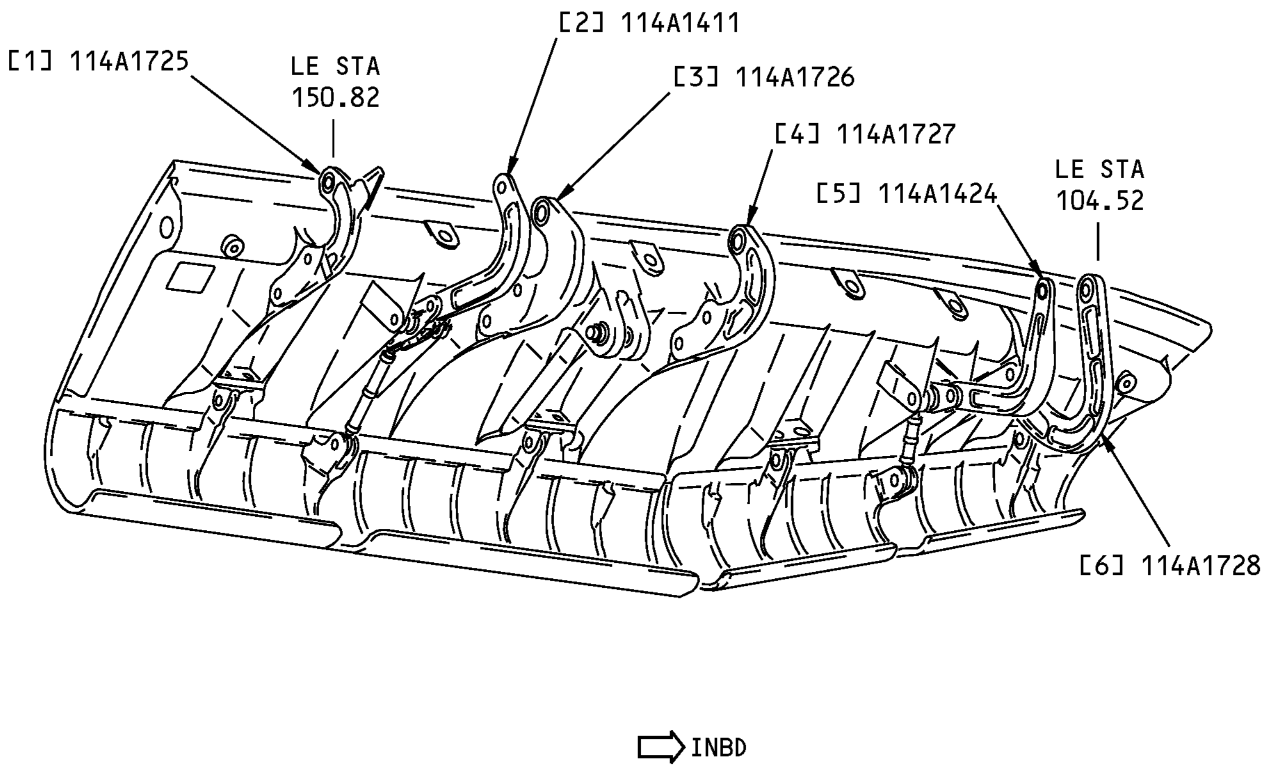
THE NUMBER 1 KRUEGER FLAP IS SHOWN,
THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

**Outboard Krueger Flap Fitting Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Hinge Fitting Number 1 | 0.60 (15.2) | Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition | |
| [2] | Hinge Fitting Numbers 2 and 3 | 0.60 (15.2) | Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition | |
| [3] | Link - Bullnose | 0.75 (19.0) | 7050-T7451 plate as given in AMS 4050. | |
| [4] | Hinge Fitting Number 4 | 0.60 (15.2) | Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition | |
| [5] | Hinge Fitting - Krueger Seal Hinge | | 17-4 PH CRES casting as given in AMS 5344, heat treated to 180 KSI minimum | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

**Inboard Krueger Flap Fitting Identification**
**Figure 3**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Hinge Fitting Number 5 | 0.60 (15.2) | Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition | |
| [2] | Link - Bullnose | 0.75 (19.0) | 7050-T7451 plate as given in AMS 4050 | |
| [3] | Hinge Fitting Number 6 | 0.80 (20.3) | Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition | |
| [4] | Hinge Fitting Number 7 | 0.80 (20.3) | Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition | |
| [5] | Link - Bullnose | 1.0 (25.4) | 7050-T7451 plate as given in AMS 4050 | |
| [6] | Hinge Fitting Number 8 | 0.70 (17.8) | Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## ALLOWABLE DAMAGE 1 - INBOARD WING LEADING EDGE (KRUEGER) FLAP FITTINGS

1. **Applicability**

   A. This subject gives the allowable damage limits for the inboard wing leading edge (Krueger) flap fittings shown in Inboard Wing Leading Edge (Krueger) Flap Fittings, Figure 101/ALLOWABLE DAMAGE 1.

2. **General**

   **WARNING:** SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH ALL FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND, OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON, OR CARBON DIOXIDE. IF WATER TOUCHES TITANIUM THAT IS ON FIRE, A STEAM EXPLOSION CAN OCCUR.

   A. Refer to SOPM 20-10-07 when you work with titanium.

   B. Remove the damaged material as necessary.

      (1) Refer to 51-10-02 for the inspection and removal of damage.

      (2) Refer to 51-30-03 for possible sources of the abrasive materials you can use to remove the damage.

      (3) Refer to 51-30-05 for possible sources of the tools and equipment you can use to remove the damage.

   C. After the damage is removed, do the steps that follow:

      **WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

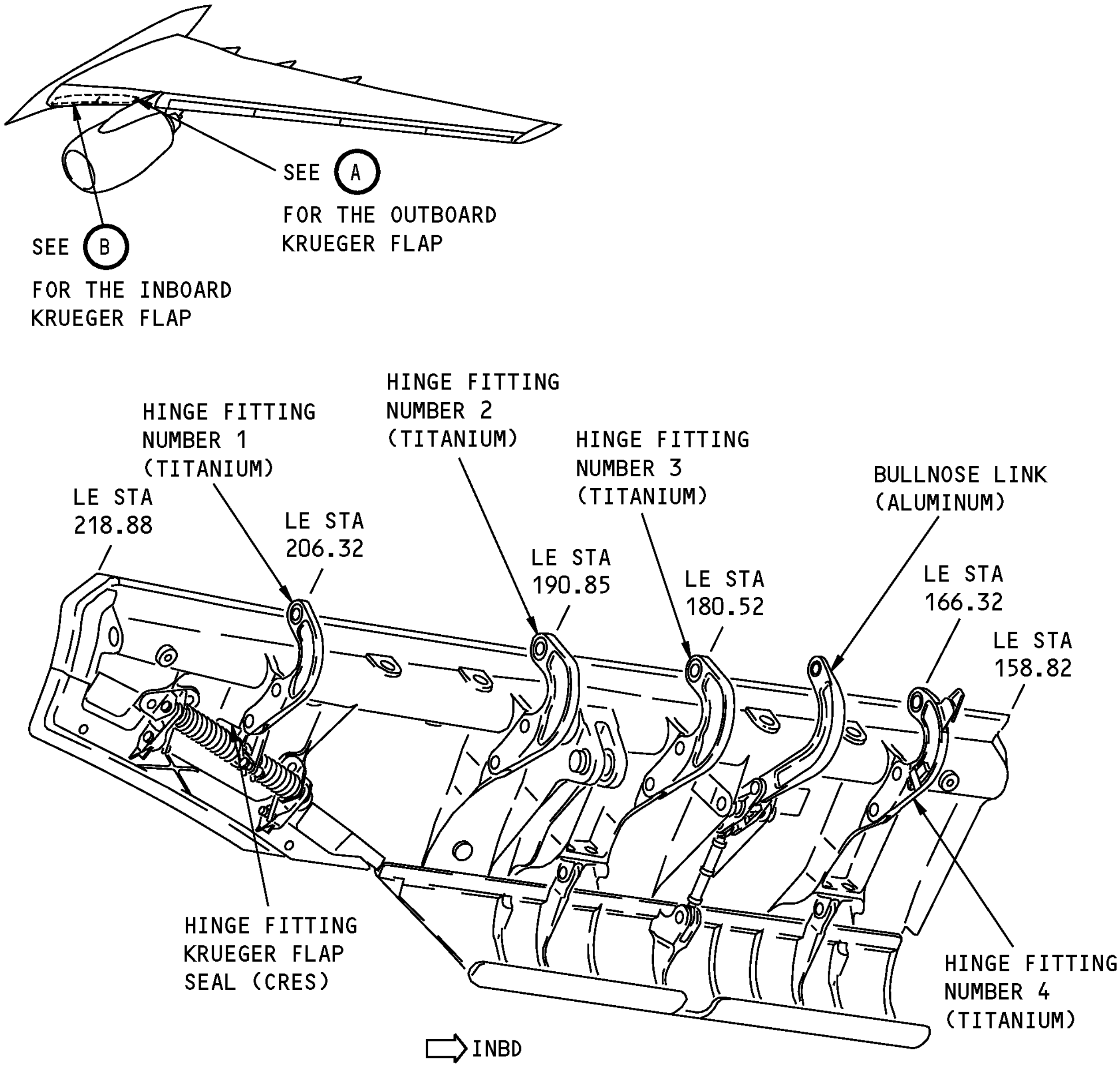
      (1) Flap peen or shot peen the reworked areas of the bullnose links, but not the inner surfaces of the link holes.

         (a) Refer to 51-20-06 for the shot peen intensity and shot number.

         (b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

      (2) Apply a chemical conversion coating to the reworked aluminum parts. Refer to 51-20-01.

      (3) Apply a layer of BMS 10-79, Type III primer to the surfaces of the reworked aluminum parts. Refer to SOPM 20-41-02.



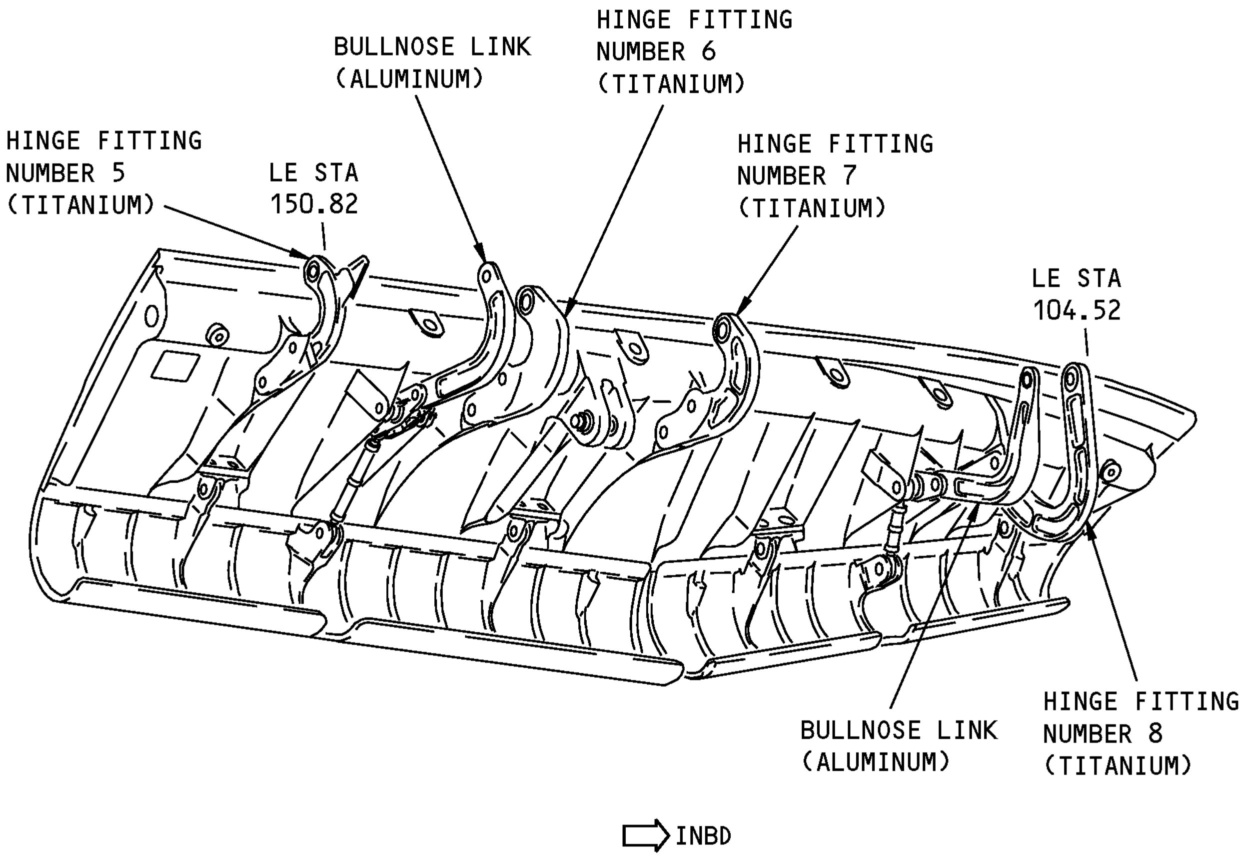

NOTE: REFER TO PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

THE NUMBER 1 KRUEGER FLAP IS SHOWN,
THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

OUTBOARD KRUEGER FLAP

A

**Inboard Wing Leading Edge (Krueger) Flap Fittings**
**Figure 101 (Sheet 1 of 2)**




NOTE: REFER TO PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

INBOARD KRUEGER FLAP

B

**Inboard Wing Leading Edge (Krueger) Flap Fittings**
**Figure 101 (Sheet 2 of 2)**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-10-07 | Machining of Titanium |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Cracks:

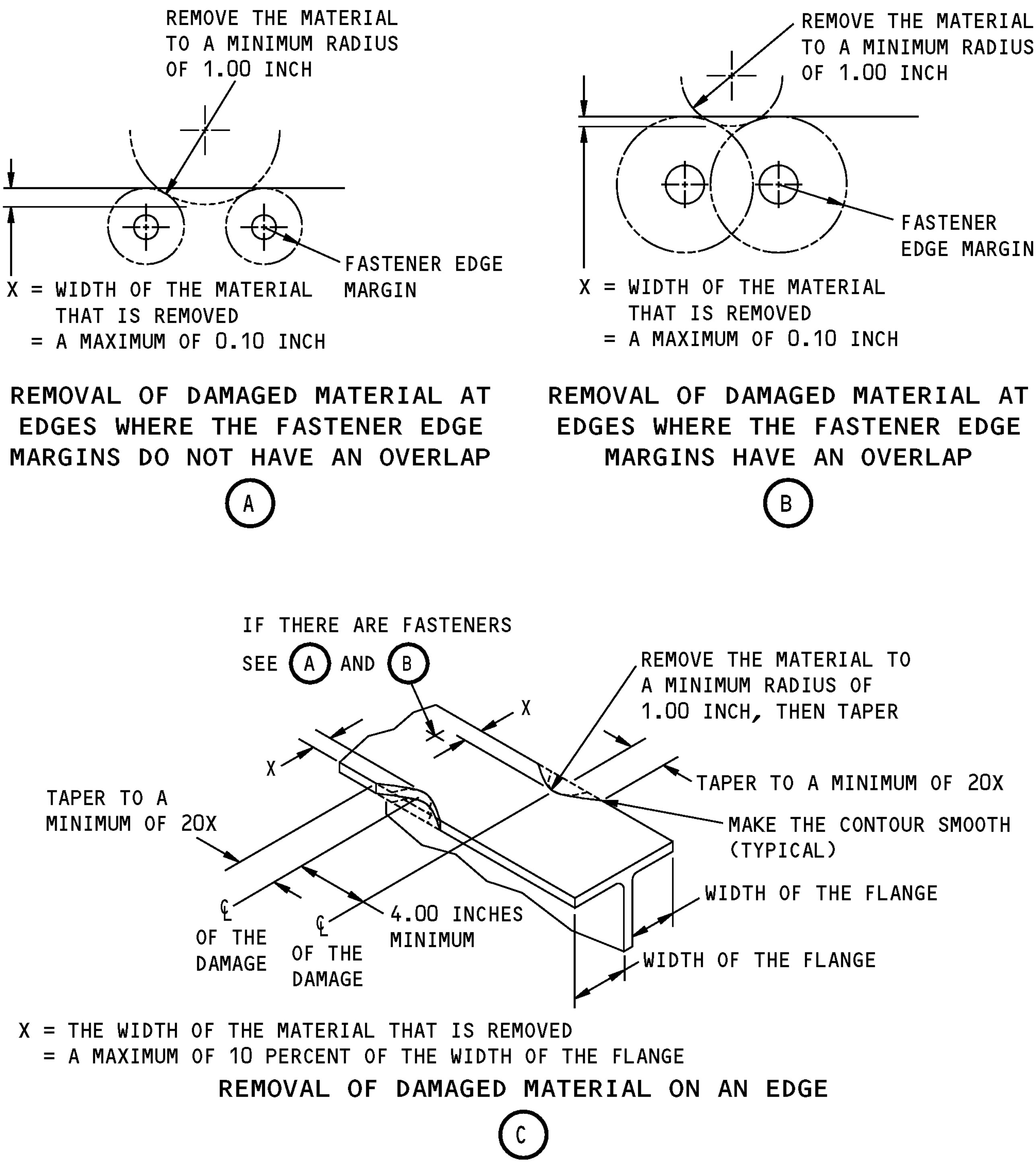
(1) Remove the damage as shown in Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , and E .

B. Nicks, Gouges, Scratches, and Corrosion:

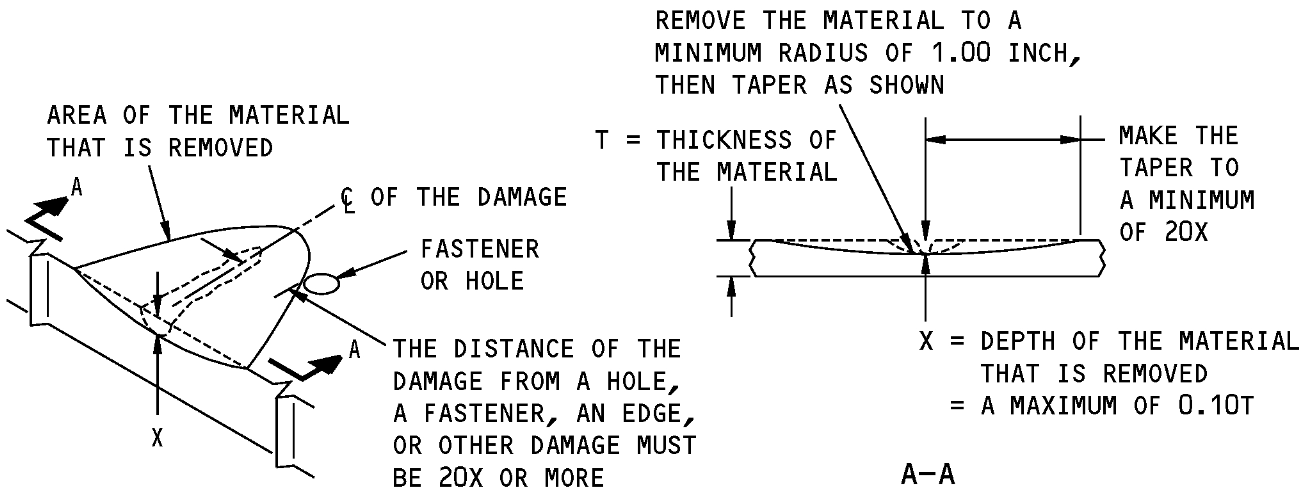
(1) Remove the damage as shown in Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage, Figure 102/ALLOWABLE DAMAGE 1, Details A through F .

C. Dents are not permitted.

D. Holes and Punctures are not permitted.

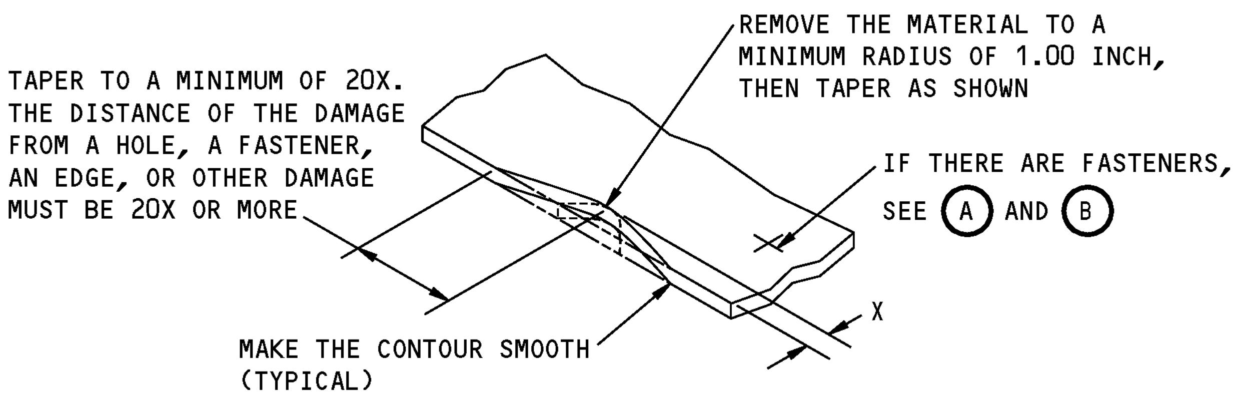



**Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage**
**Figure 102 (Sheet 1 of 3)**




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

D

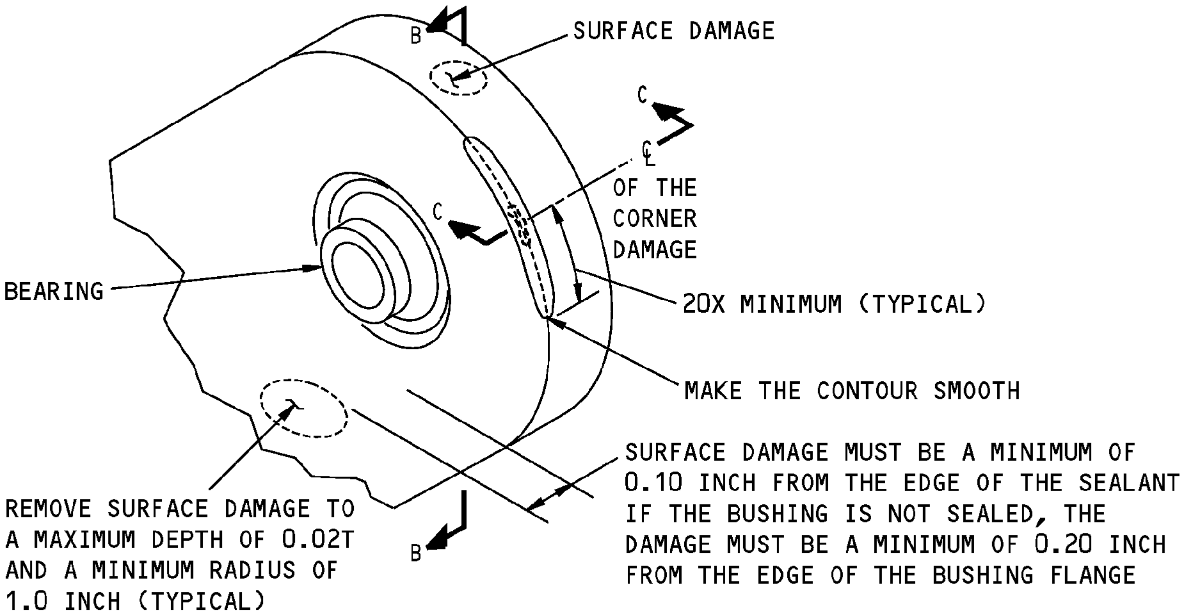



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

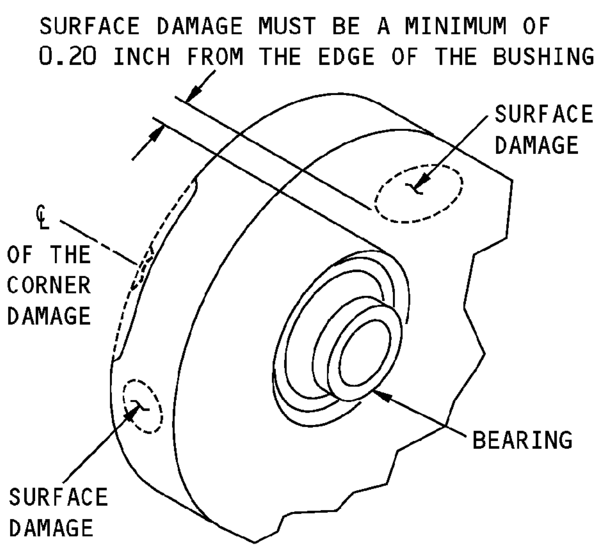
E

**Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage**
**Figure 102 (Sheet 2 of 3)**

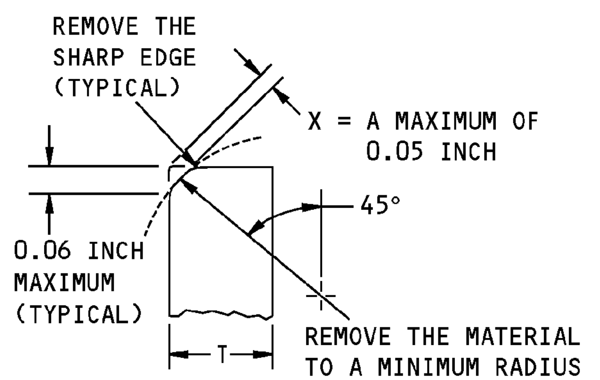



REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BEARING

F



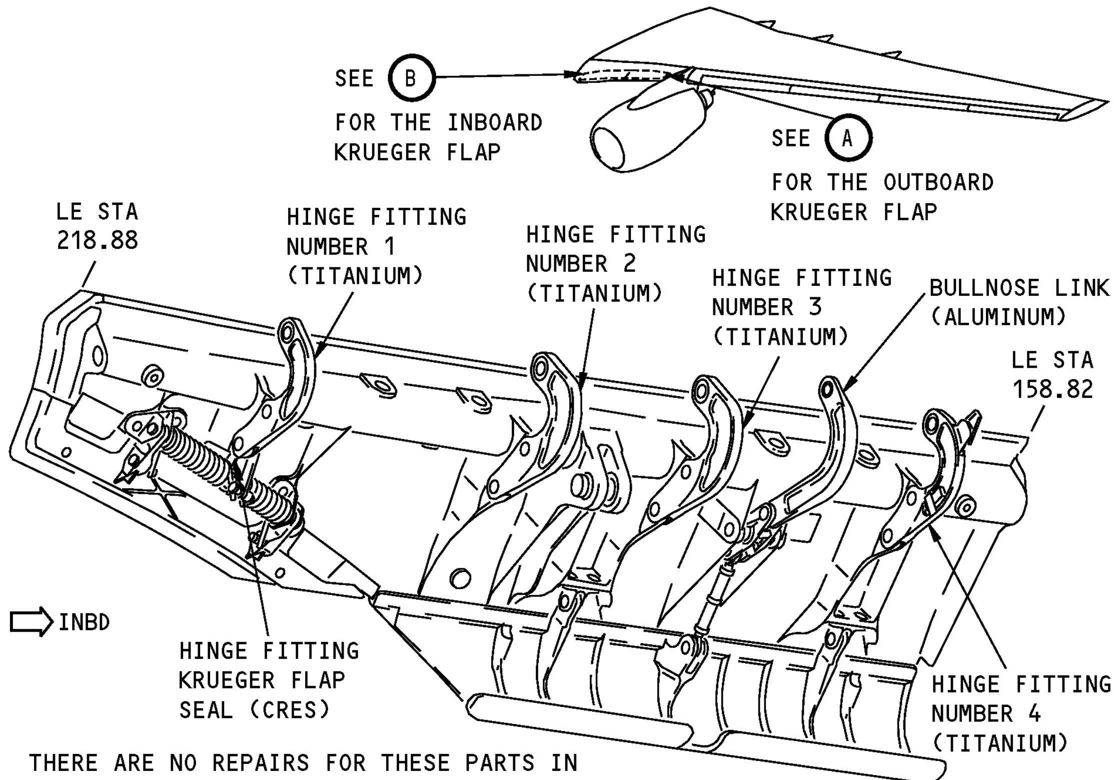

B-B




C-C

**Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage**
**Figure 102 (Sheet 3 of 3)**

## REPAIR 1 - INBOARD WING LEADING EDGE (KRUEGER) FLAP FITTINGS



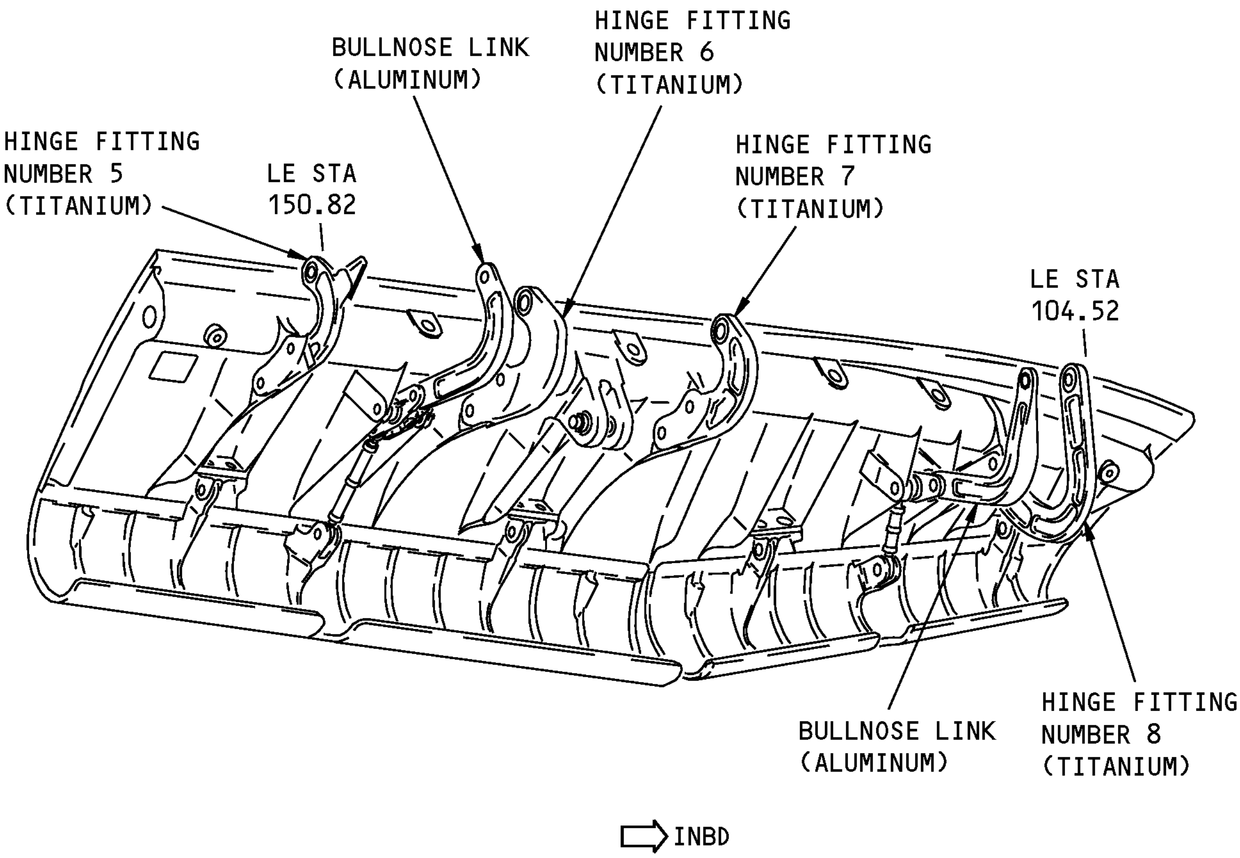

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

THE NUMBER 1 KRUEGER FLAP IS SHOWN,
THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

OUTBOARD KRUEGER FLAP

A

**Inboard Wing Leading Edge (Krueger) Flap Fitting Repair**
**Figure 201 (Sheet 1 of 2)**




NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

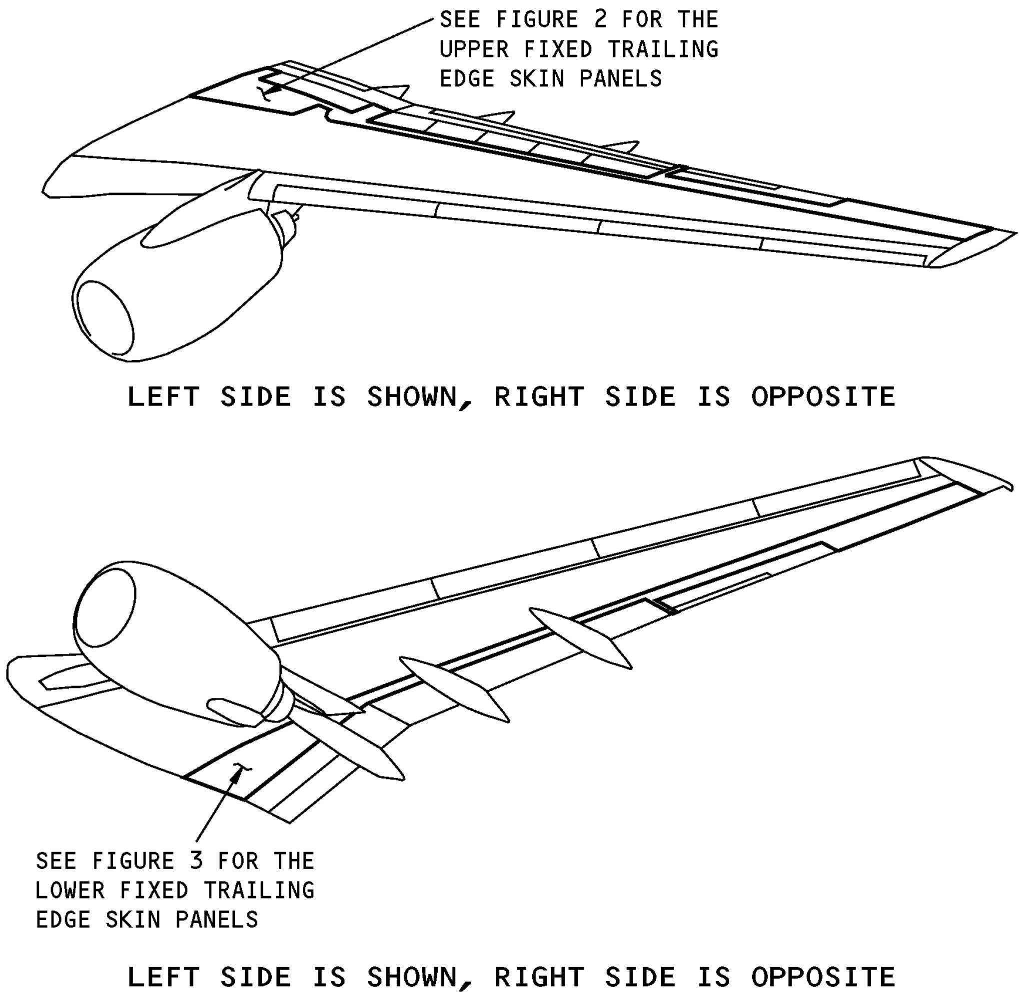
THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

INBOARD KRUEGER FLAP

B

**Inboard Wing Leading Edge (Krueger) Flap Fitting Repair**
**Figure 201 (Sheet 2 of 2)**

## IDENTIFICATION 1 - WING FIXED TRAILING EDGE SKIN PANELS




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

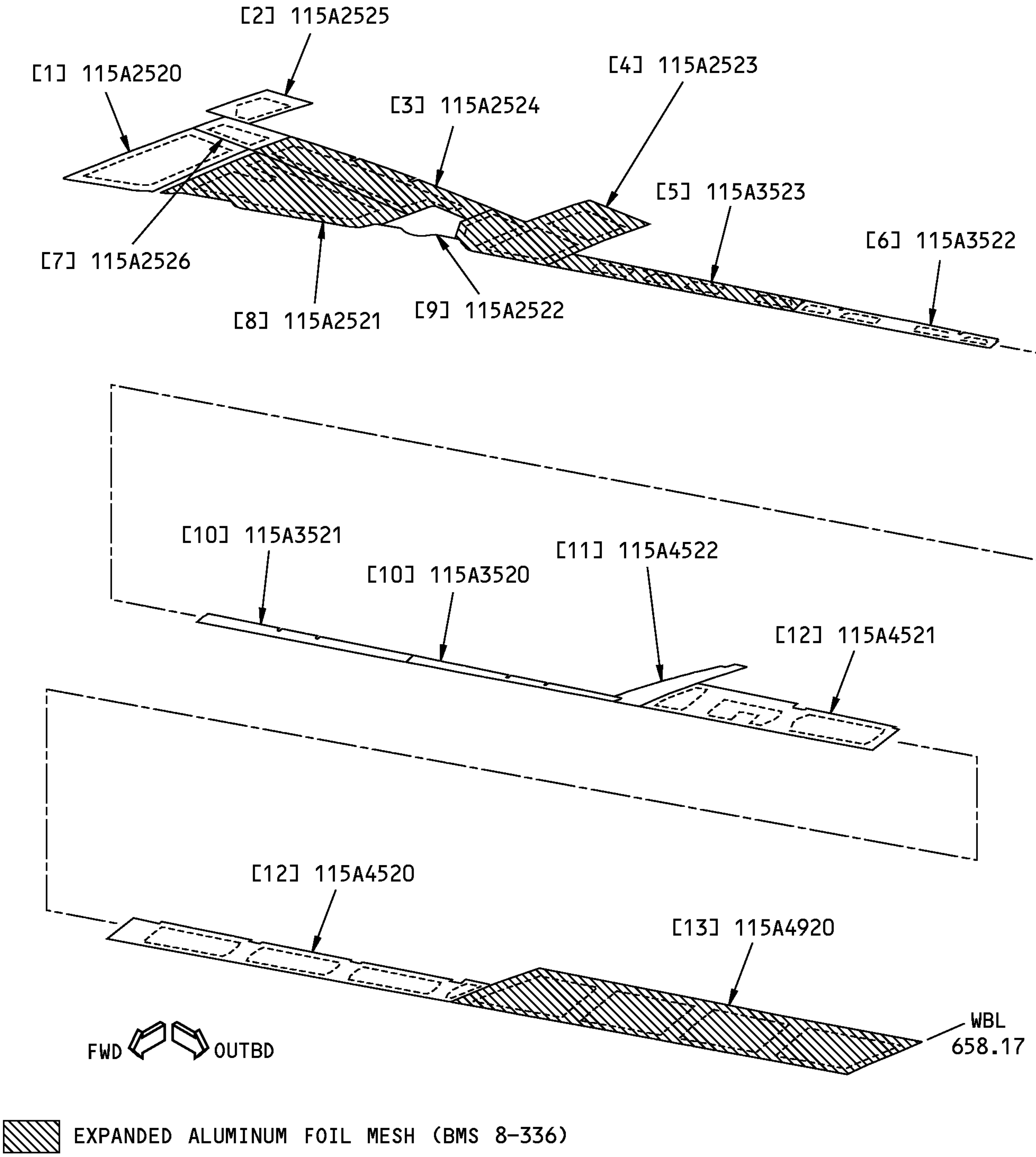
**Wing Fixed Trailing Edge Skin Panel Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 115A0001 | Functional Product Collector, Fixed Trailing Edge - Left Wing |
| 115A0002 | Functional Product Collector, Fixed Trailing Edge - Right Wing |
| 115A0005 | Functional Product Collector, Fixed Trailing Edge to Final Assembly |
| 115A2000 | Upper Installation - Inboard Fixed Trailing Edge |
| 115A2001 | Lower Installation - Inboard Fixed Trailing Edge |
| 115A2200 | Boltcover Installation - Inboard Fixed Trailing Edge |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 115A2500 | Panel Installation - Inboard Fixed Trailing Edge |
| 115A2510 | Panel Assembly - Upper Inboard Fixed Trailing Edge |
| 115A2511 | Panel Assembly - Upper, Inboard Fixed Trailing Edge |
| 115A2512 | Panel Assembly - Upper Inboard Fixed Trailing Edge |
| 115A2513 | Panel Assembly - Upper Inboard Fixed Trailing Edge |
| 115A2514 | Panel Assembly - Upper Inboard Fixed Trailing Edge |
| 115A2515 | Panel Assembly - Upper Inboard Fixed Trailing Edge |
| 115A2525 | Panel Assembly - Bonded Part, Upper Inboard Fixed Trailing Edge |
| 115A2710 | Panel Assembly - Lower Inboard Fixed Trailing Edge |
| 115A2711 | Panel Assembly - Lower Inboard Fixed Trailing Edge |
| 115A2712 | Panel Assembly - Lower Inboard Fixed Trailing Edge |
| 115A2713 | Panel Assembly - Lower Inboard Fixed Trailing Edge |
| 115A2714 | Panel Assembly - Lower Inboard Fixed Trailing Edge |
| 115A2715 | Panel Assembly - Lower Inboard Fixed Trailing Edge |
| 115A2716 | Panel Assembly - Lower Inboard Fixed Trailing Edge |
| 115A3500 | Panel Installation - Mid Span Fixed Trailing Edge, Upper |
| 115A3510 | Panel Assembly - Mid Span Fixed Trailing Edge, Upper |
| 115A3511 | Panel Assembly - Mid Span Fixed Trailing Edge, Upper |
| 115A3512 | Panel Assembly - Mid Span Fixed Trailing Edge, Upper |
| 115A3513 | Panel Assembly - Mid Span Fixed Trailing Edge, Upper |
| 115A3700 | Panel Installation - Mid Span Fixed Trailing Edge, Lower |
| 115A3710 | Panel Assembly - Mid Span Fixed Trailing Edge, Lower |
| 115A3711 | Panel Assembly - Mid Span Fixed Trailing Edge, Lower |
| 115A3712 | Panel Assembly - Mid Span Fixed Trailing Edge, Lower |
| 115A4500 | Panel Installation - Outboard Foxed Trailing Edge, Upper |
| 115A4510 | Panel Assembly - Outboard Fixed Trailing Edge, Upper |
| 115A4511 | Panel Assembly - Outboard Fixed Trailing Edge, Upper |
| 115A4512 | Panel Assembly - Outboard Fixed Trailing Edge, Upper |
| 115A4700 | Panel Installation - Outboard Foxed Trailing Edge, Lower |
| 115A4710 | Panel Assembly - Outboard Fixed Trailing Edge, Lower |
| 115A4711 | Panel Assembly - Outboard Fixed Trailing Edge, Lower |
| 115A4712 | Panel Assembly - Outboard Fixed Trailing Edge, Lower |
| 115A4714 | Panel Assembly - Outboard Fixed Trailing Edge, Lower |
| 115A4900 | Structure Installation - Panels and Supports, Outboard Fixed Trailing Edge |
| 115A4901 | Structure Assembly - Panels and Supports, Outboard Fixed Trailing Edge |
| 115A4910 | Panel Assembly - Outboard Fixed Trailing Edge, Upper |
| 115A4911 | Panel Assembly - Outboard Fixed Trailing Edge, Lower |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 115A4912 | Panel Assembly - Outboard Fixed Trailing Edge, Lower |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

UPPER SURFACE

**Wing Upper Fixed Trailing Edge Skin Panels**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Panel Assembly - Bonded Part | | Glass fiber reinforced plastic (GFRP) honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 1.00 (25.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [2] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220*[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781)*[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [3] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336 Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [4] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [5] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 0.30 (7.6) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [6] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [7] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |

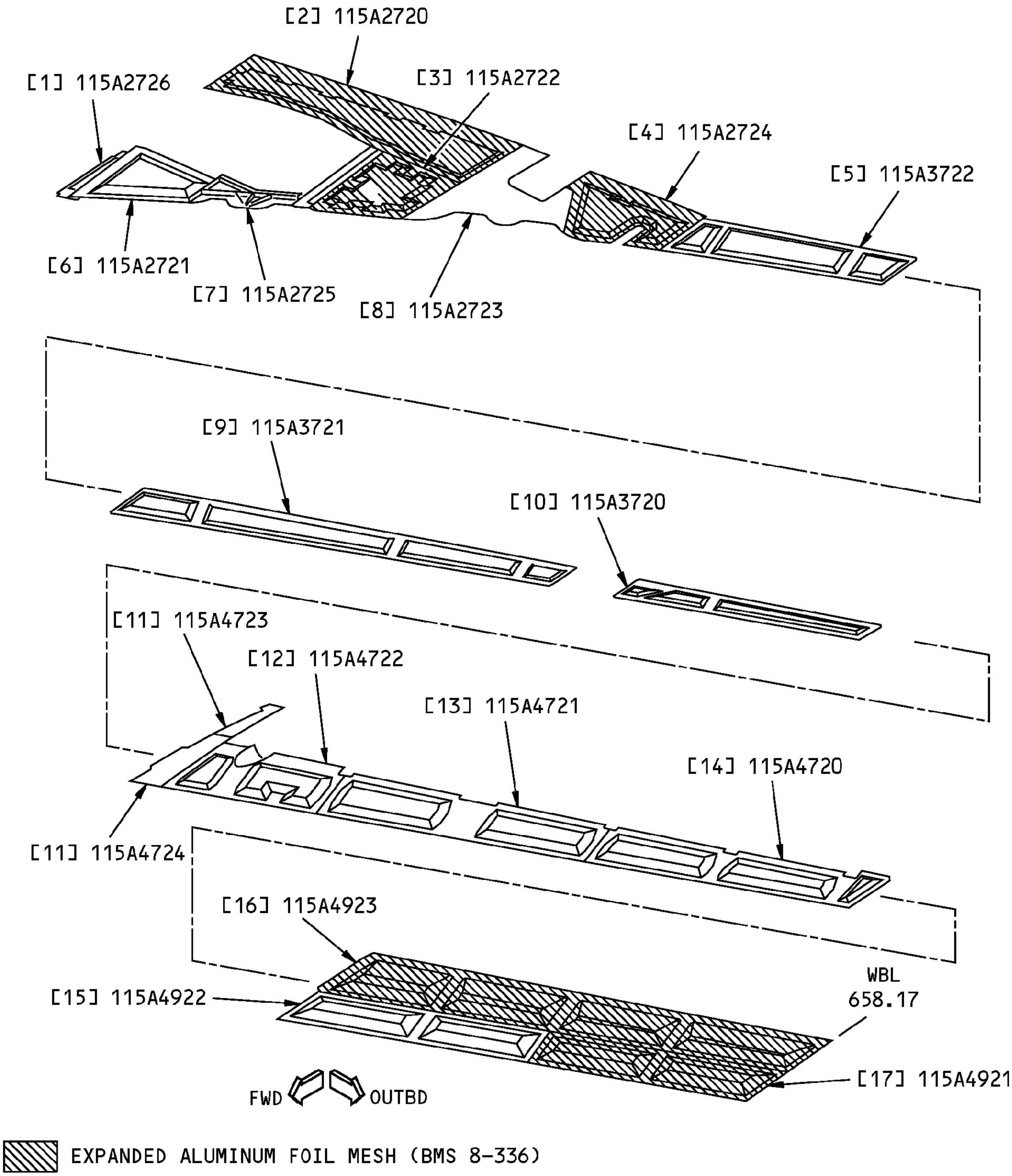
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [8] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 1.00 (25.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [9] | Panel Assembly | 0.10 (2.5) | 7075-T62 sheet as given in QQ-A-250/12 | |
| [10] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| [11] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 778 *[2] | |
| [12] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]<br>Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [13] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 1.00 (25.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

*[2] Refer to the production drawings for the ply lay-up and orientation.

*[3] Refer to the production drawings for the core ribbon direction.




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

LOWER SURFACE

**Wing Lower Fixed Trailing Edge Panels**
**Figure 3**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Panel Assembly - Bonded Part | | Glass fiber reinforced plastic (GFRP) honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [2] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [3] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [4] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [5] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 7781 or 1581 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [6] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.80 (20.3) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [7] | Panel Assembly | | A356-T6 Casting as given in AMS 2418 | |
| [8] | Panel Assembly | 0.10 (2.5) | 2024-T42 sheet | |
| [9] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | For cum line numbers 475 and on and airplanes that include Service Bulletin 737-57-1255 |
| | Skin Ply (3) | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies (3) | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120, 220, 1581, or 7781 *[2] | |
| | Core (5) | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T *[1] | MATERIAL | EFFECTIVITY |
| [9] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | For all other airplanes |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2] | |
| | Core | 0.30 (7.6)<br>0.40 (10.2)<br>0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [10] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.30 (7.6) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [11] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| [12] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2] | |

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7)<br>0.80 (20.3) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [13] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 7781 or 1581 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.80 (20.3) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [14] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.50 (12.7)<br>0.80 (20.3) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 Type V, Grade 3.0 *[3] | |
| [15] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 7781 or 1581 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 1.00 (25.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [16] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 0.90 (22.9)<br>1.00 (25.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |
| [17] | Panel Assembly - Bonded Part | | GFRP honeycomb sandwich | |
| | Lightning Protection Ply | | Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016 | |
| | Skin Ply | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Doubler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2] | |
| | Filler Plies | | Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or 1581 (Optional: 7781) *[2] | |
| | Core | 1.00 (25.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3] | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

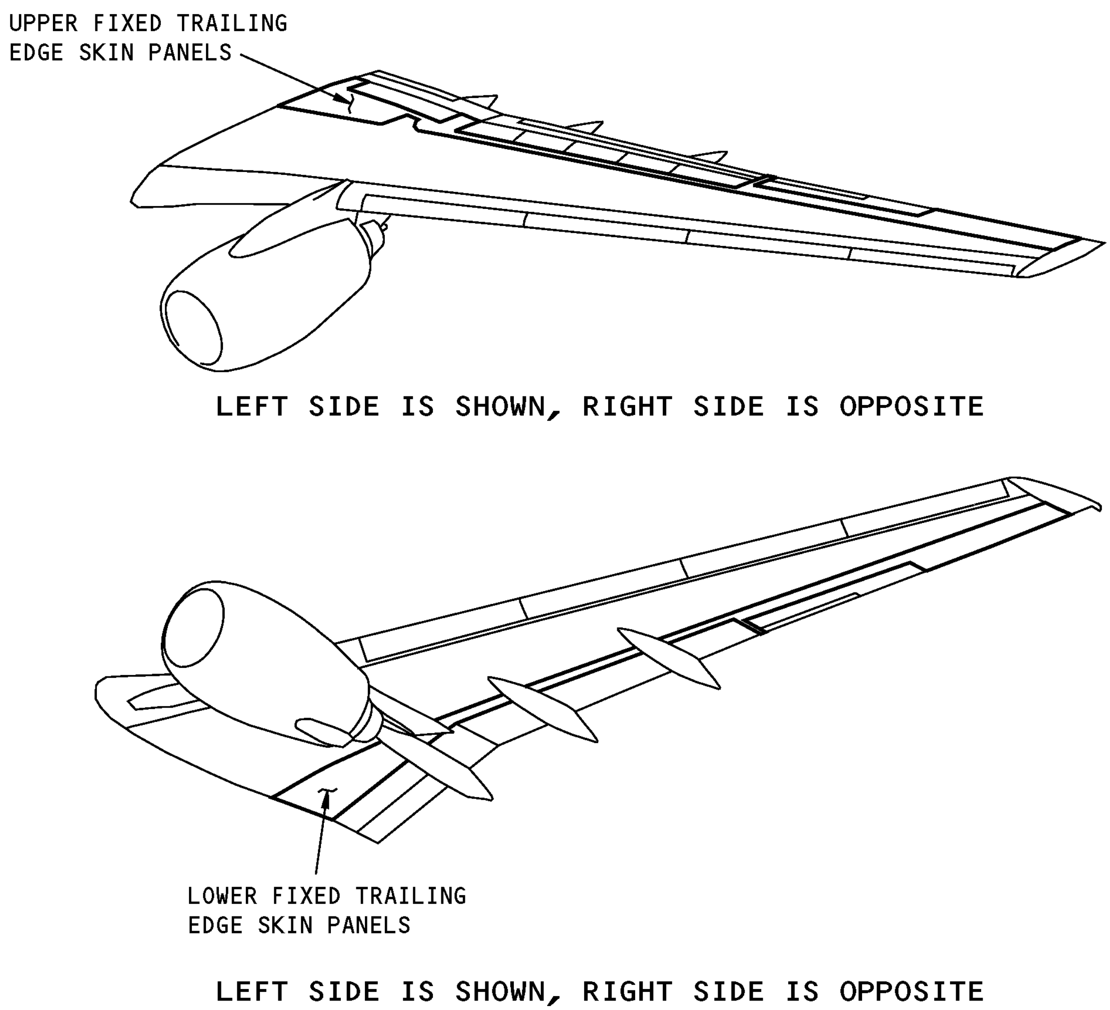
*[2] Refer to the production drawings for the ply lay-up and orientation.

*[3] Refer to the production drawings for the core ribbon direction.

## ALLOWABLE DAMAGE 1 - WING FIXED TRAILING EDGE SKIN PANELS

1. **Applicability**

   A. Allowable Damage 1 is applicable to damage on the wing fixed trailing edge skin panels as shown in Wing Fixed Trailing Edge Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.




**Wing Fixed Trailing Edge Skin Panel Location**
**Figure 101**

2. **General**

   A. Do the steps that follow for the skin panels made of Glass Fabric Reinforced Plastic (GFRP).

      (1) Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

      **NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator, can be used.

(a) For the honeycomb core areas, the tap test is an alternative procedure to an instrumented NDT.

(2) Remove all the contamination and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and the cleanup procedures.

(a) Refer to 51-70-04 for the damage removal procedures.

(b) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(3) Refer to Definitions of the Damage Size, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(4) Refer to Definitions of the Facesheets, Figure 103/ALLOWABLE DAMAGE 1 for the definitions of the facesheets of a honeycomb core area.

(5) Refer to Wing Fixed Trailing Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the GFRP panels.

(6) Some GFRP panels have BMS 8-336 expanded aluminum foil mesh as shown in Wing Fixed Trailing Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1. If damage occurs to the expanded aluminum foil mesh, do the steps that follow:

(a) Refer to 51-70-14 for the allowable damage limits for the expanded aluminum foil mesh.

(b) Seal the damaged area as given in 51-70-14.

(7) Seal all damaged areas that do not have BMS 8-336 expanded aluminum foil mesh with the steps that follow.

(a) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

1) Make a temporary seal.

a) Apply aluminum foil tape (speed tape).

b) Keep a record of the location.

c) Make sure the tape is in satisfactory condition at each 400 flight hour interval or more frequently.

d) Seal the damage permanently no later than 5000 flight hours from the time the seal was made.

2) Make a permanent seal.

a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given 51-70-08.

b) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to SOPM 20-44-04.

c) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(b) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

1) Use a vacuum and heat to remove moisture from the solid laminate or the honeycomb cells. Refer to 51-70-04.

2) Make a temporary seal with aluminum foil tape (speed tape).

3) Keep a record of the location.

4) Repair the damage no later than 5000 flight hours from the time the seal was made.

B. Do the steps that follow for the skin panels made of formed aluminum sheet.

(1) Refer to Wing Fixed Trailing Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the aluminum panels.

(2) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of the abrasive and other materials you need to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

(3) After you remove the damage, do the steps that follow:

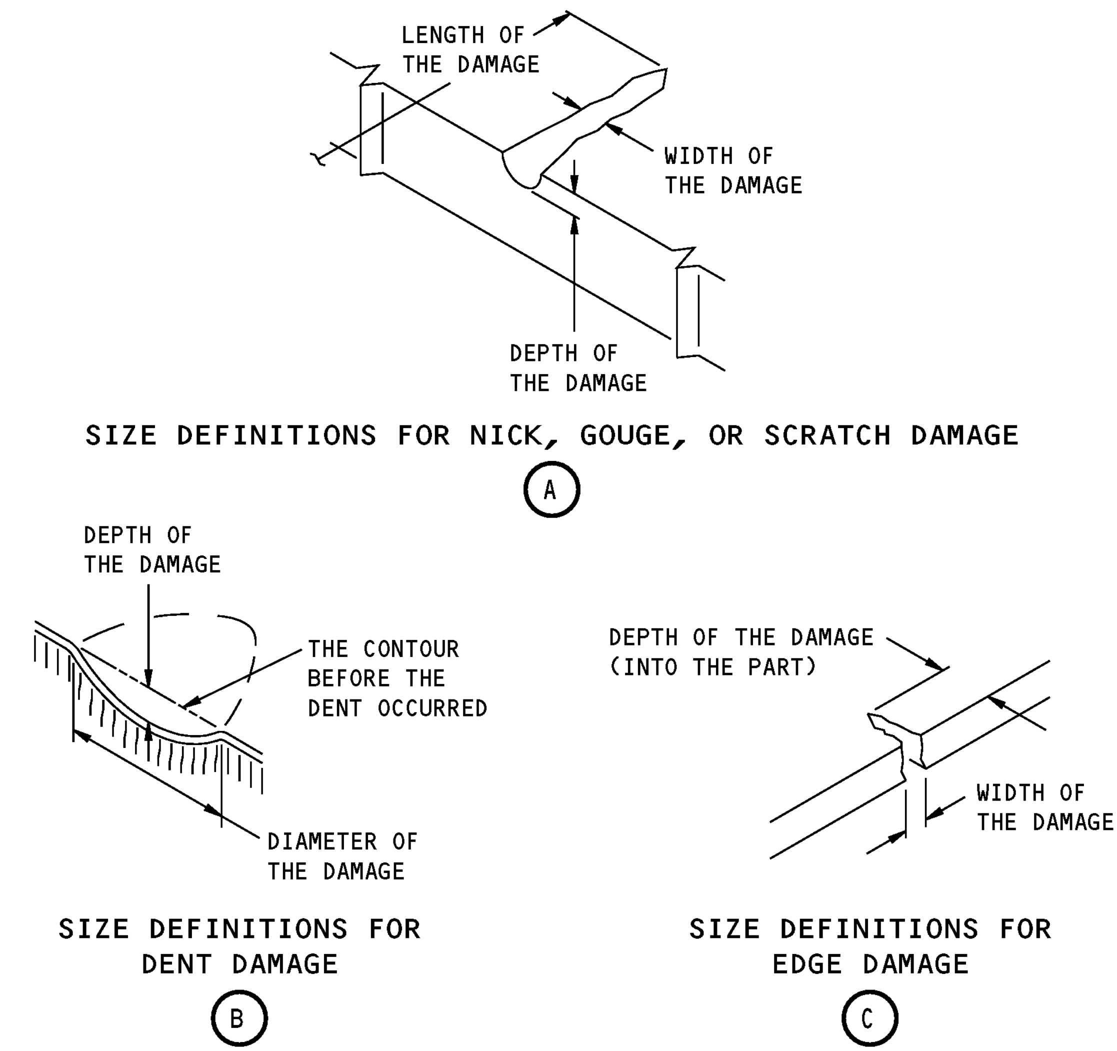
(a) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(b) Apply one layer of BMS 10-11, Type I primer to the conversion coated, reworked areas. Refer to SOPM 20-41-02.

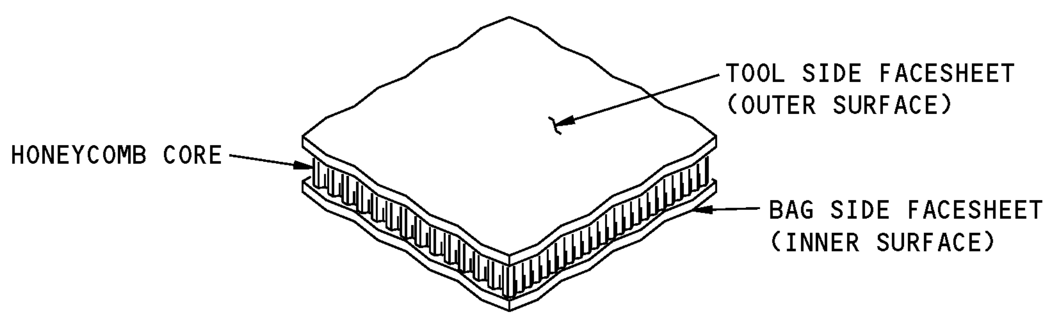
(c) Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the economic performance of the airplane. Refer to 51-10-01.

**Table 101**:

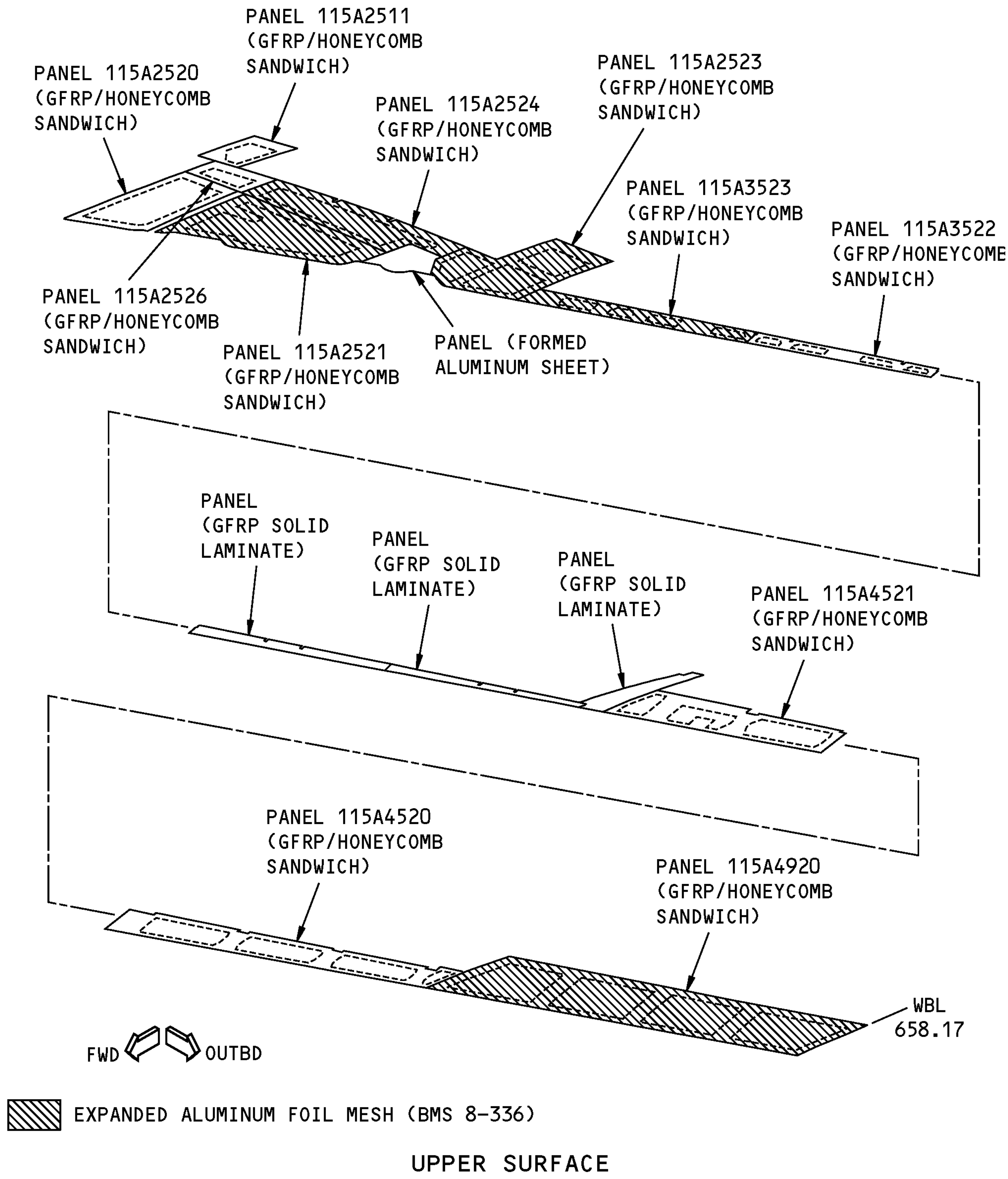
| **PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS** | |
|---|---|
| TYPE OF STRUCTURE | PARAGRAPH |
| GFRP/HONEYCOMB SANDWICH PANELS - HONEYCOMB CORE AREAS | 4.A |
| GFRP/HONEYCOMB SANDWICH PANELS - SOLID LAMINATE AREAS | 4.B |
| GFRP SOLID LAMINATE PANELS | 4.C |
| FORMED ALUMINUM SHEET PANELS | 4.D |
| ALUMINUM CASTING PANELS | 4.E |



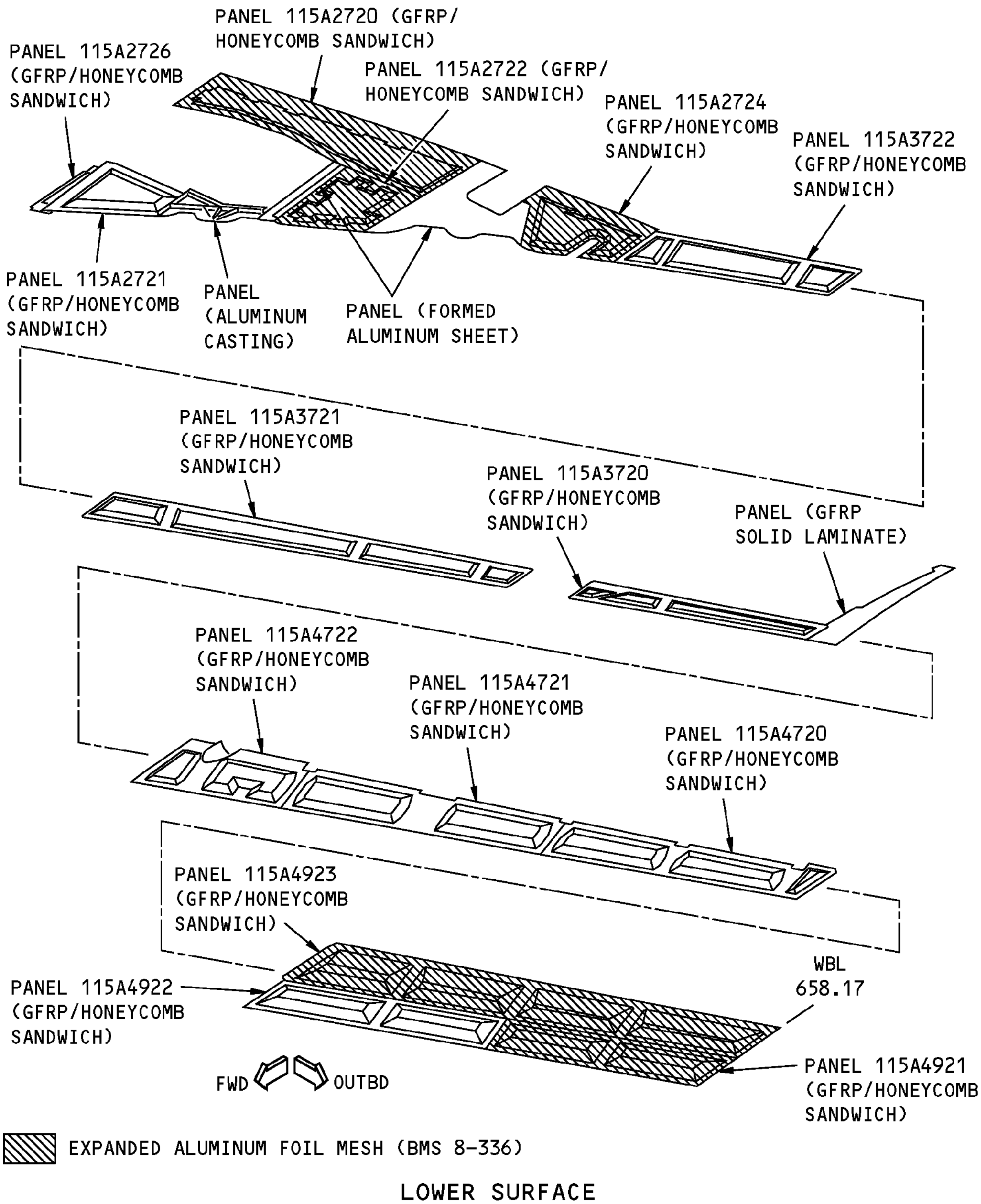

**Definitions of the Damage Size**
**Figure 102**




**Definitions of the Facesheets**
**Figure 103**




**Wing Fixed Trailing Edge Skin Allowable Damage**
**Figure 104 (Sheet 1 of 2)**




**Wing Fixed Trailing Edge Skin Allowable Damage**
**Figure 104 (Sheet 2 of 2)**

**3. References**

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| 51-70-14 | STRUCTURES WITH ALUMINUM COATINGS AND FOILS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

**4. Allowable Damage Limits**

A. GFRP/Honeycomb Sandwich Panels - Honeycomb Core Areas

(1) Nicks, gouges and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges and scratches that cause damage to the glass fibers are permitted if:

(a) The depth of the damage is less than the limits given in Table 102/ALLOWABLE DAMAGE 1 and Table 103/ALLOWABLE DAMAGE 1.

**NOTE**: Use the limits for holes and punctures if the depth of the damage is more than the limits given in Table 102/ALLOWABLE DAMAGE 1 and Table 103/ALLOWABLE DAMAGE 1.

(b) They are a maximum of 2.0 inches in length

(c) They are a maximum of 0.25 inch in width

(d) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

**Table 102**:

| MAXIMUM DEPTH OF NICK, GOUGE, OR SCRATCH DAMAGE FOR GFRP/HONEYCOMB SANDWICH CORE AREAS ON THE WING UPPER SURFACE | | |
|---|---|---|
| PANEL DRAWING NUMBER | MAXIMUM NUMBER OF PLIES IN DEPTH | |
| | UPPER SURFACE (TOOL SIDE FACESHEET) | LOWER SURFACE (BAG SIDE FACESHEET) |
| 115A2511 | 2 | 2 |
| 115A2520 | 1 | 1 |

| MAXIMUM DEPTH OF NICK, GOUGE, OR SCRATCH DAMAGE FOR GFRP/HONEYCOMB SANDWICH CORE AREAS ON THE WING UPPER SURFACE | | |
|---|---|---|
| PANEL DRAWING NUMBER | MAXIMUM NUMBER OF PLIES IN DEPTH | |
| | UPPER SURFACE (TOOL SIDE FACESHEET) | LOWER SURFACE (BAG SIDE FACESHEET) |
| 115A2521 | 1 | 1 |
| 115A2523 | 1 | Fiber damage is not permitted |
| 115A2524 | 1 | 1 |
| 115A2526 | 2 | Fiber damage is not permitted |
| 115A3522 | 2 | 1 |
| 115A3523 | 2 | 1 |
| 115A4520 | 2 | Fiber damage is not permitted |
| 115A4521 | 2 | Fiber damage is not permitted |
| 115A4920 | 2 | Fiber damage is not permitted |

**Table 103**:

| MAXIMUM DEPTH OF NICK, GOUGE, OR SCRATCH DAMAGE FOR GFRP/HONEYCOMB SANDWICH CORE AREAS ON THE WING LOWER SURFACE | | |
|---|---|---|
| PANEL DRAWING NUMBER | MAXIMUM NUMBER OF PLIES IN DEPTH | |
| | UPPER SURFACE (BAG SIDE FACESHEET) | LOWER SURFACE (TOOL SIDE FACESHEET) |
| 115A2720 | 1 | 2 |
| 115A2721 | 1 | 2 |
| 115A2722 | 1 | 2 |
| 115A2724 | 1 | 2 |
| 115A2726 | 1 | 2 |
| 115A3720 | 1 | 1 |
| 115A3721 | 1 | 1 |
| 115A3722 | 1 | 1 |
| 115A4720 | Fiber damage is not permitted | Fiber damage is not permitted |
| 115A4721 | Fiber damage is not permitted | Fiber damage is not permitted |
| 115A4722 | Fiber damage is not permitted | 1 |
| 115A4921 | 1 | 1 |
| 115A4922 | 1 | 1 |
| 115A4923 | 1 | Fiber damage is not permitted |

Table 104:

| MAXIMUM DAMAGE DIMENSIONS AND MINIMUM SPACING DIMENSIONS FOR DENT, HOLE AND PUNCTURE, AND DELAMINATION DAMAGE ON GFRP PANEL HONEYCOMB CORE AREAS ON THE WING UPPER SURFACE | | | | |
|---|---|---|---|---|
| PANEL DRAWING NUMBER | UPPER SURFACE (TOOL SIDE FACESHEET) | | LOWER SURFACE (BAG SIDE FACESHEET) | |
| | MAXIMUM DAMAGE DIMENSION "D" (INCHES) | MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES) | MAXIMUM DAMAGE DIMENSION "D" (INCHES) | MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES) |
| 115A2511 | 1.5 | (1.5)x(D + d) | 1.5 | (1.5)x(D + d) |
| 115A2520 | 2.0 | (3.0)x(D + d) | 2.0 | (2.5)x(D + d) |
| 115A2521 | 1.5 | (3.5)x(D + d) | 2.0 | (2.0)x(D + d) |
| 115A2523 | 1.0 | (4.5)x(D + d) | 1.0 | (5.0)x(D + d) |
| 115A2524 | 1.5 | (2.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A2526 | 2.0 | (1.5)x(D + d) | 1.0 | (2.5)x(D + d) |
| 115A3522 | 1.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A3523 | 1.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A4520 | 1.5 | (1.5)x(D + d) | 1.0 | (1.5)x(D + d) |
| 115A4521 | 1.5 | (1.5)x(D + d) | 0.75 | (4.0)x(D + d) |
| 115A4920 | 2.0 | (1.5)x(D + d) | 1.5 | (2.5)x(D + d) |

Table 105:

| MAXIMUM DAMAGE DIMENSIONS AND MINIMUM SPACING DIMENSIONS FOR DENT, HOLE AND PUNCTURE, AND DELAMINATION DAMAGE ON GFRP PANEL HONEYCOMB CORE AREAS ON THE WING LOWER SURFACE | | | | |
|---|---|---|---|---|
| PANEL DRAWING NUMBER | UPPER SURFACE (BAG SIDE FACESHEET) | | LOWER SURFACE (TOOL SIDE FACESHEET) | |
| | MAXIMUM DAMAGE DIMENSION "D" (INCHES) | MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES) | MAXIMUM DAMAGE DIMENSION "D" (INCHES) | MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES) |
| 115A2720 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A2721 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A2722 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A2724 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A2726 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A3720 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A3721 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A3722 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A4720 | 1.0 | (2.0)x(D + d) | 1.5 | (1.5)x(D + d) |
| 115A4721 | 1.0 | (2.0)x(D + d) | 1.5 | (1.5)x(D + d) |
| 115A4722 | 1.0 | (2.0)x(D + d) | 1.5 | (1.5)x(D + d) |
| 115A4921 | 2.0 | (1.5)x(D + d) | 2.0 | (1.5)x(D + d) |
| 115A4922 | 2.0 | (1.5)x(D + d) | 1.5 | (1.5)x(D + d) |
| 115A4923 | 2.0 | (1.5)x(D + d) | 1.5 | (2.5)x(D + d) |

(3) Dents are permitted if:

(a) The dimension of the damage is less than the limits given in Table 104/ALLOWABLE DAMAGE 1 and Table 105/ALLOWABLE DAMAGE 1.

(b) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if:

(a) They are a maximum of one facesheet and the core in depth

(b) The dimension of the damage is less than the limits given in Table 104/ALLOWABLE DAMAGE 1 and Table 105/ALLOWABLE DAMAGE 1.

(c) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if:

(a) The dimension of the damage is less than the limits given in Table 104/ALLOWABLE DAMAGE 1 and Table 105/ALLOWABLE DAMAGE 1.

(b) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Pararaph 2.

B. GFRP/Honeycomb Sandwich Panels - Solid Laminate Areas

(1) Nicks, gouges and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges and scratches that cause damage to the glass fibers are not permitted.

(3) Dents are permitted if they are:

**NOTE**: Use the limits for holes and punctures if there is glass fiber damage.

(a) A maximum of 0.50 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if:

(a) They are a maximum of 0.25 inch in diameter

(b) They are a minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, the edge of a hole, or the edge of the material. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

(b) A maximum of 0.50 inch in width

(c) A minimum of 2.5D (D = the diameter of the damage) from the edge of other damage or the edge of a hole. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(7) Edge erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ALLOWABLE DAMAGE 1.

C. GFRP Solid Laminate Panels (does not include the solid laminate areas in honeycomb sandwich panels)

(1) Nicks, gouges and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges and scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of 1 ply in depth

**NOTE**: Use the limits for holes and punctures if the depth of the damage is more than 1 ply in depth.

(b) A maximum of 3.0 inches in length

(c) A maximum of 0.25 inch in width

(d) A minimum of 0.50 inch away from the edge of other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 0.5 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch in diameter

(b) A minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if:

(a) They are a maximum of 0.5 inch in diameter

(b) They are a minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, the edge of a hole, or the edge of the material. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

(b) A maximum of 0.50 inch in width

(c) They are a minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, the edge of a hole, or the edge of the material. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(7) Edge erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ALLOWABLE DAMAGE 1.

D. Panels Made of Formed Aluminum Sheet

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , and C.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E.

(3) Dents:

(a) Dents are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail F

2) There is 1.5 square inches or less of dent area for each 144 square inches of panel area.

(b) If the depth of a dent is larger than 0.050 inch and the dent agrees with the conditions shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail F , do the steps that follow:

1) Fill or rework the dent as given in 51-70-01

2) If you fill the dent, then do the steps that follow:

a) Seal the damage with aluminum foil tape (speed tape)

b) Make an inspection of the dent at each 400 flight hour interval or more frequently.

- Make sure the tape is in satisfactory condition
- Repair the dent if the damage becomes larger.

c) Repair the damage at or before 1000 flight hours have occurred.

(4) Holes and Punctures are permitted if:

(a) They are 0.25 inch in diameter or less

(b) The edge of the damage is a minimum of 1.0 inch away from a fastener hole, an edge, or other damage

(c) They are filled with a 2017-T3 or 2117-T4 aluminum flush head rivet. Install the rivet without sealant.

E. Aluminum Casting Panel

(1) Cracks:

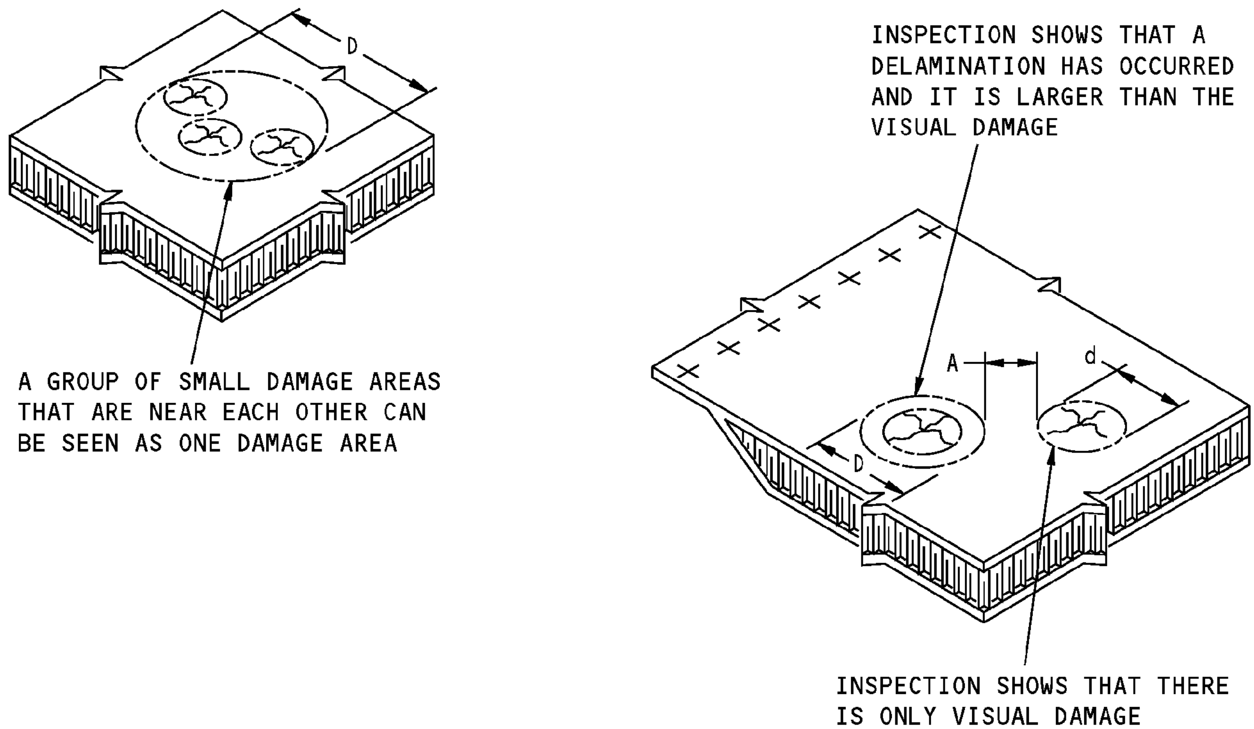
(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , and C.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



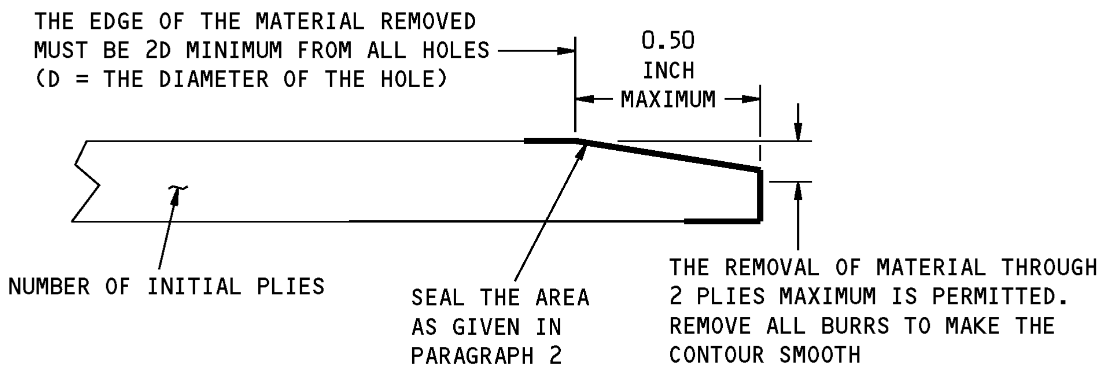

NOTE: THE DIMENSION OF A DAMAGE AREA IS EITHER THE DIMENSION OF THE VISUAL DAMAGE OR THE DIMENSION OF THE DELAMINATION. USE THE DIMENSION OF THE LARGER DAMAGE.

D IS THE LARGER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

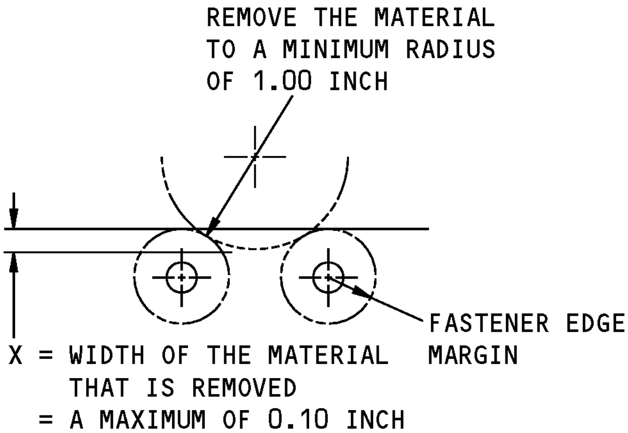
d IS THE SMALLER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

A IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

**Damage that is Permitted to Honeycomb Core Areas**
**Figure 105**

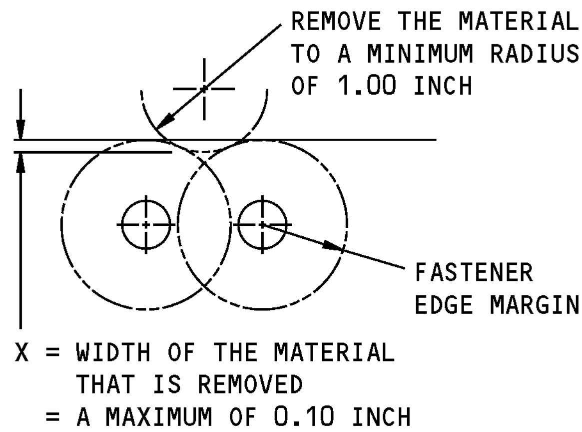



**Cleanup and Sealing of Edge Erosion**
**Figure 106**



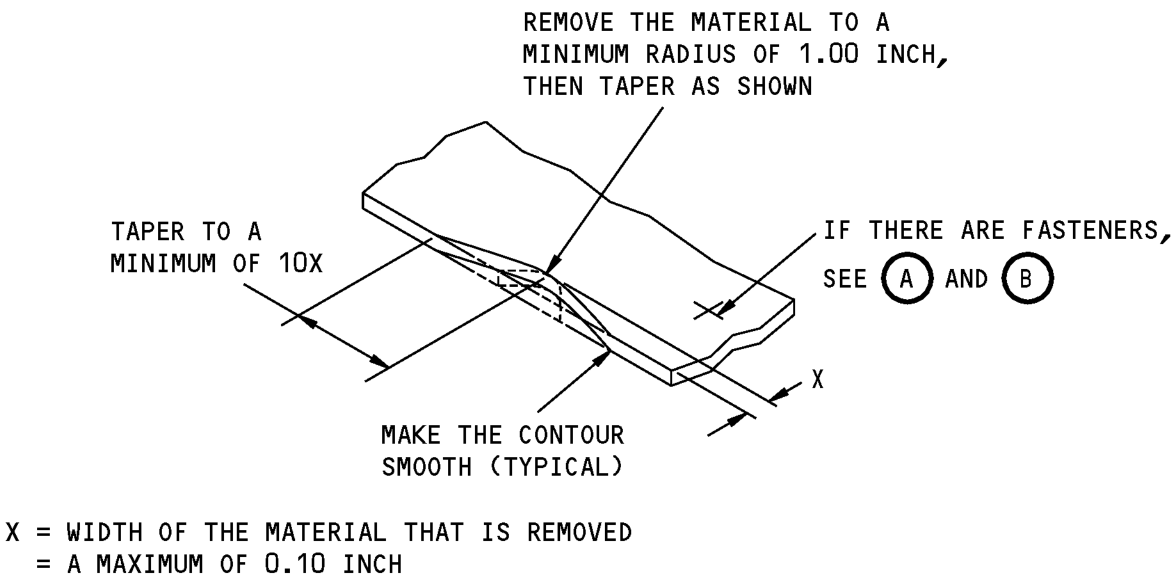

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

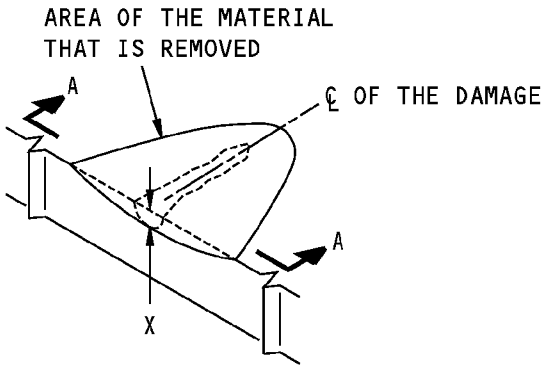
B




REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

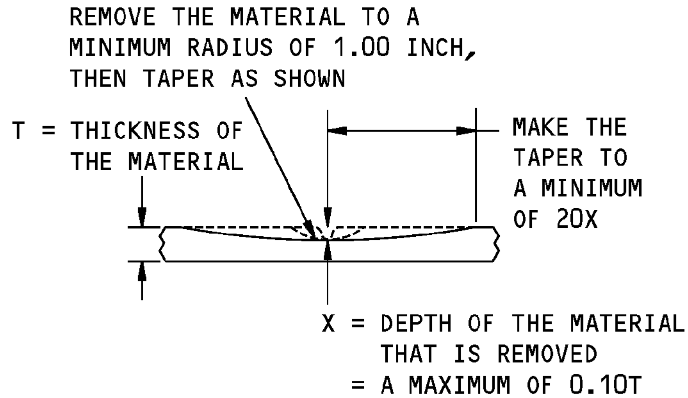
C

**Allowable Damage Limits**
**Figure 107 (Sheet 1 of 3)**

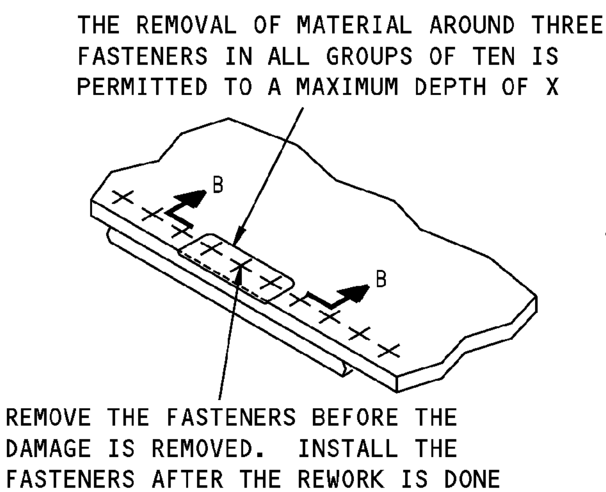



REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

D

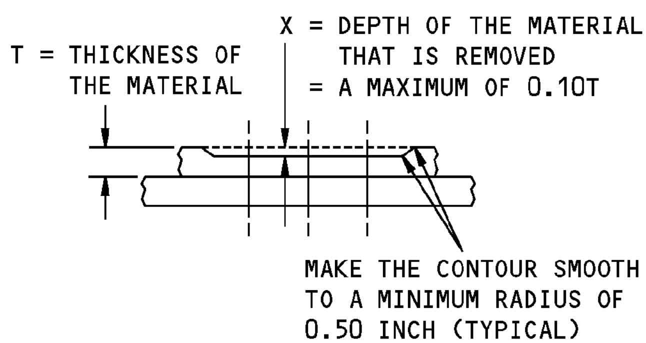



A–A



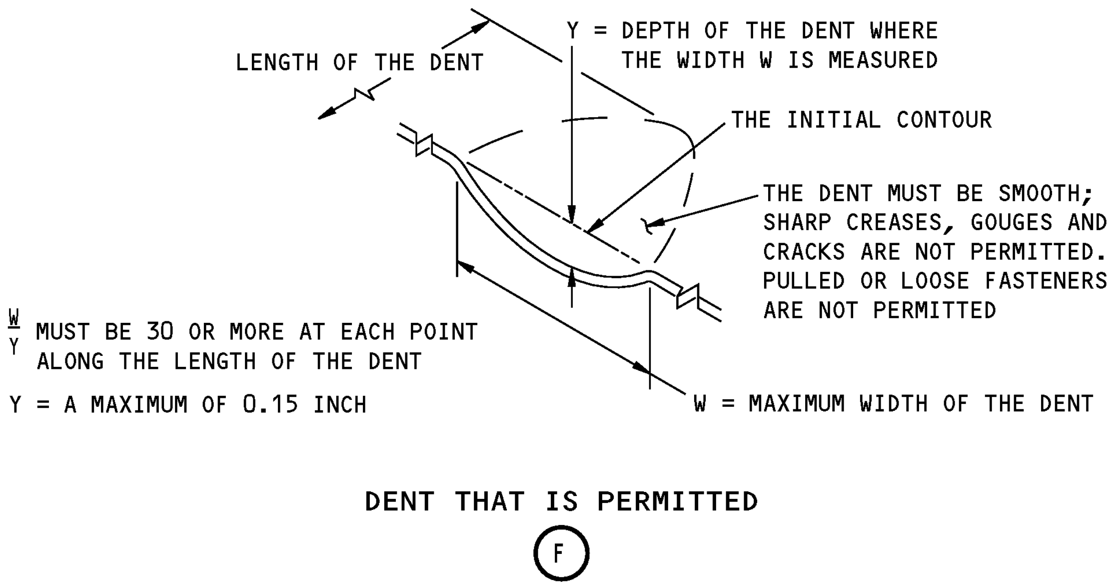

REMOVAL OF DAMAGE AROUND THE
FASTENERS ON AN EDGE
OR A SURFACE

E




B–B

**Allowable Damage Limits**
**Figure 107 (Sheet 2 of 3)**



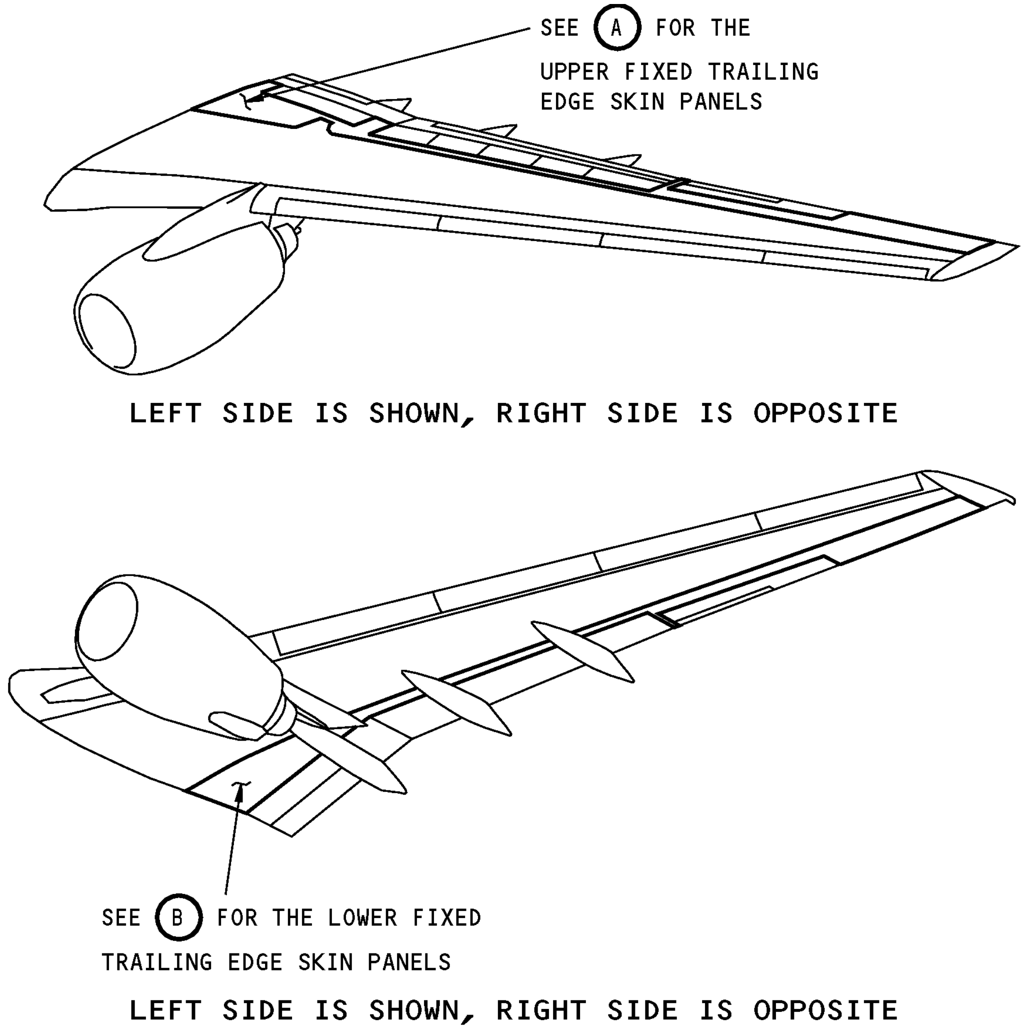

**Allowable Damage Limits**
**Figure 107 (Sheet 3 of 3)**

## REPAIR 1 - WING FIXED TRAILING EDGE SKIN PANELS MADE OF COMPOSITE MATERIALS

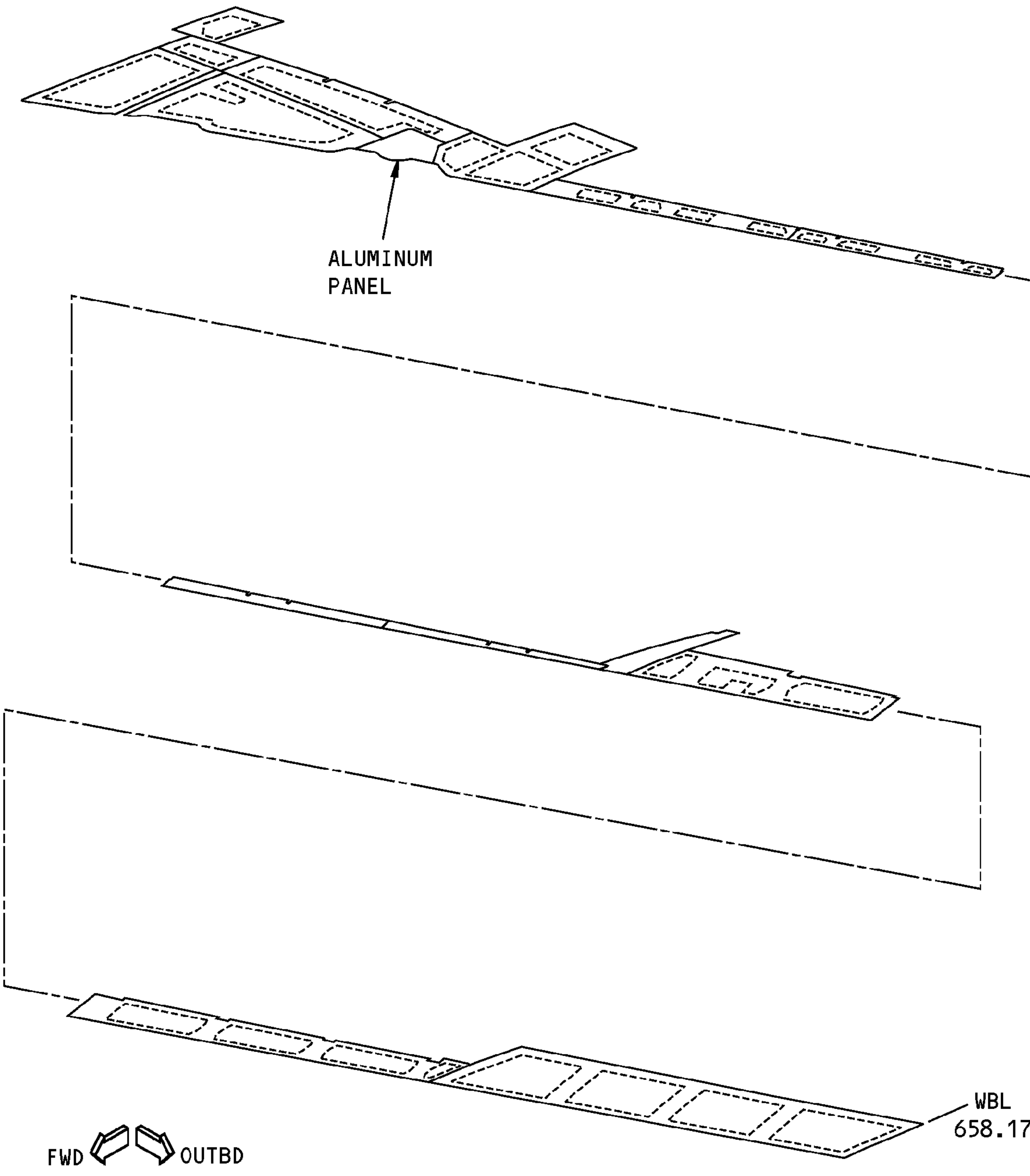
### 1. Applicability

A. Repair 1 is applicable to damage to skin panels made of Glass Fiber Reinforced Plastic (GFRP) as shown in Wing Fixed Trailing Edge Skin Panel Location, Figure 201/REPAIR 1.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.




**Wing Fixed Trailing Edge Skin Panel Location**
**Figure 201 (Sheet 1 of 3)**

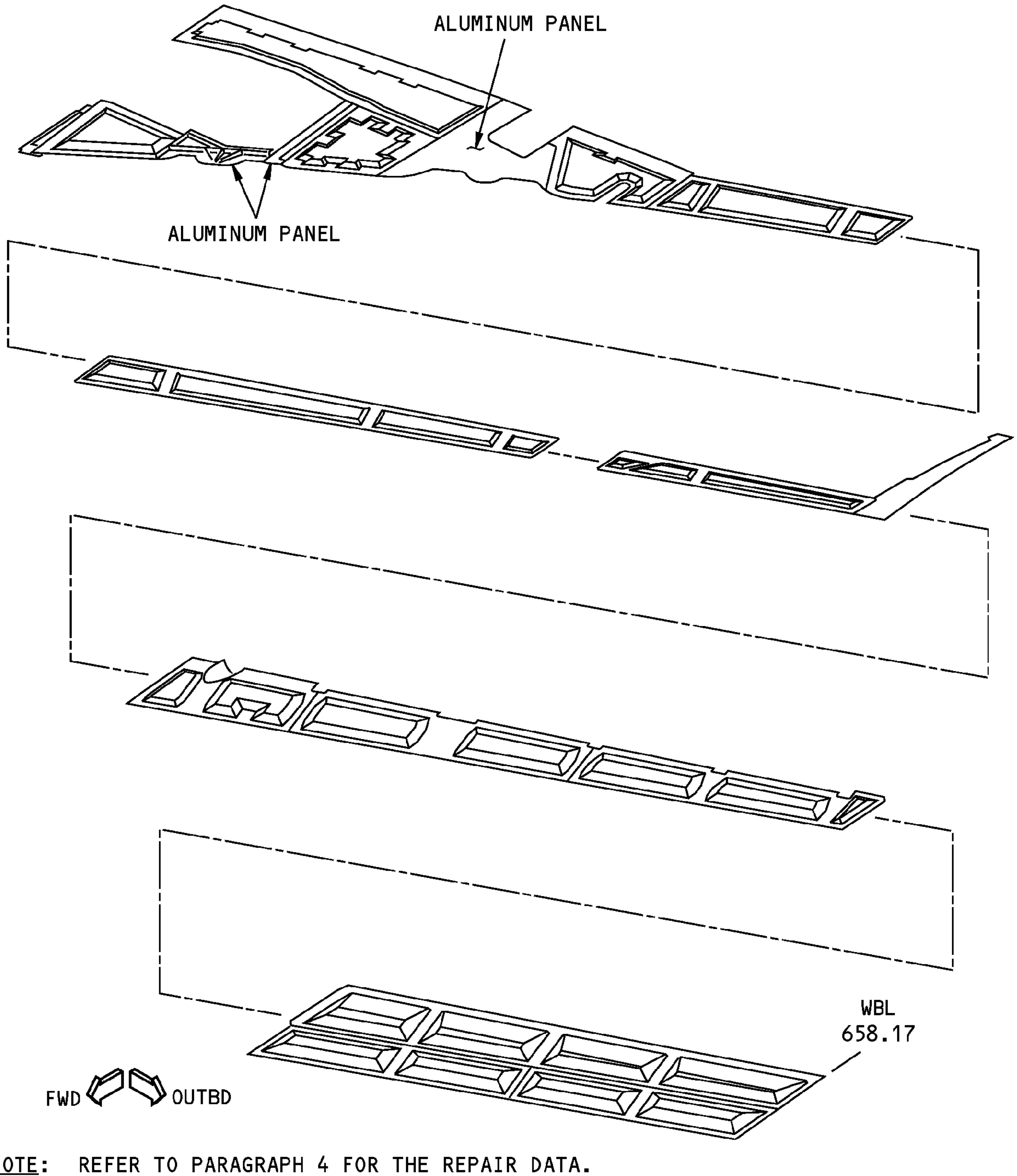



NOTE: REFER TO PARAGRAPH 4 FOR THE REPAIR DATA.

UPPER SURFACE

A

**Wing Fixed Trailing Edge Skin Panel Location**
**Figure 201 (Sheet 2 of 3)**




NOTE: REFER TO PARAGRAPH 4 FOR THE REPAIR DATA.

LOWER SURFACE

B

**Wing Fixed Trailing Edge Skin Panel Location**
**Figure 201 (Sheet 3 of 3)**

2. **General**

A. Repair 1 gives repair instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the panels as necessary to get access to the inner surface of the skin.

(1) Remove the necessary fasteners. Refer to 51-40-02 for information on fastener removal.

(2) If a fastener hole is damaged, refer to 51-70-04 or 51-70-05, as applicable.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-01 for the inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Definitions of the Damage Size, Figure 202/REPAIR 1 for the definitions of the length, width, and depth of damage.

(2) Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

D. Some trailing edge panels have expanded aluminum foil mesh (BMS 8-336). If damage occurs:

(1) Refer to 51-70-14 before you repair a facesheet that has expanded aluminum foil mesh (BMS 8-336).

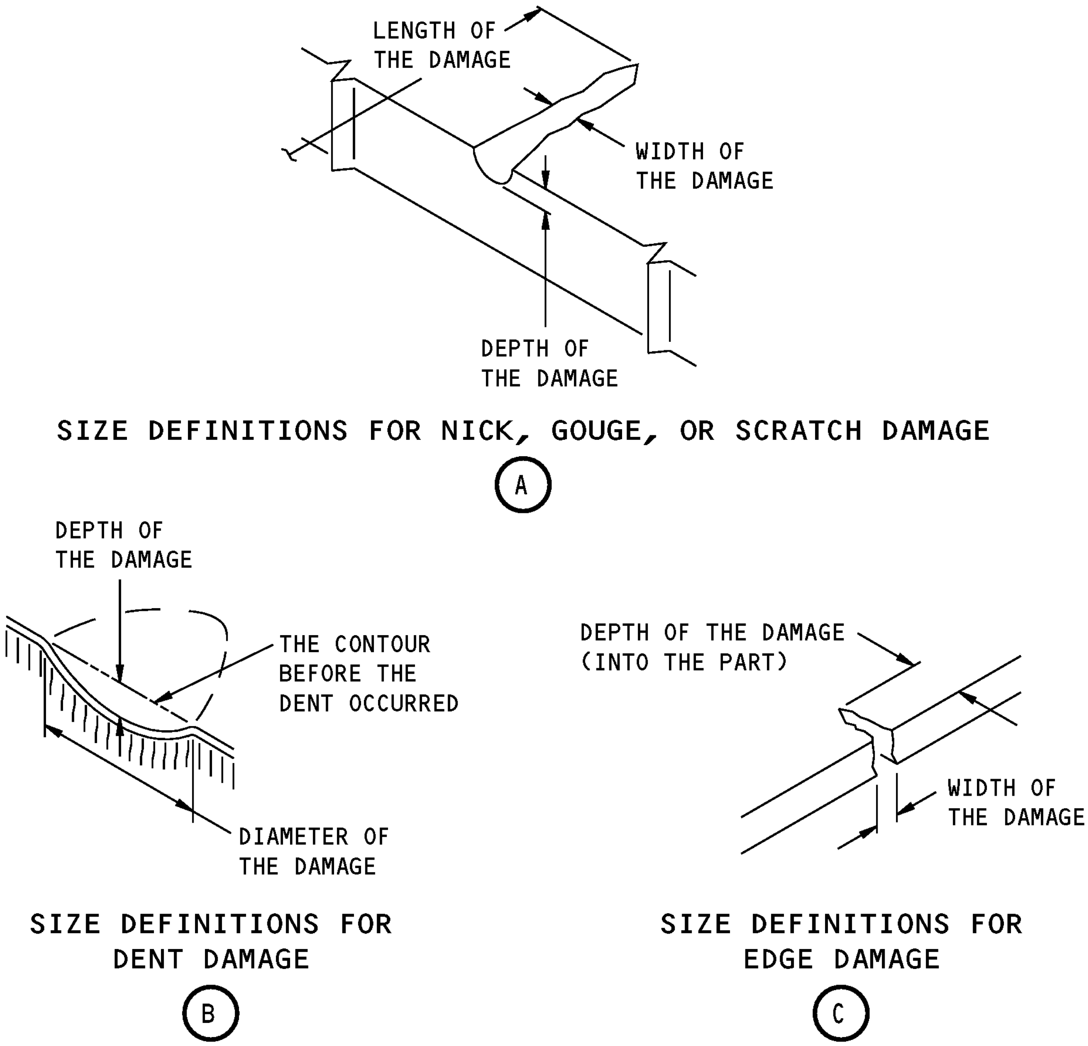
(2) Refer to 51-70-14 after you repair a facesheet that has expanded aluminum foil mesh (BMS 8-336).

E. Do the repair as given in Paragraph 4./REPAIR 1

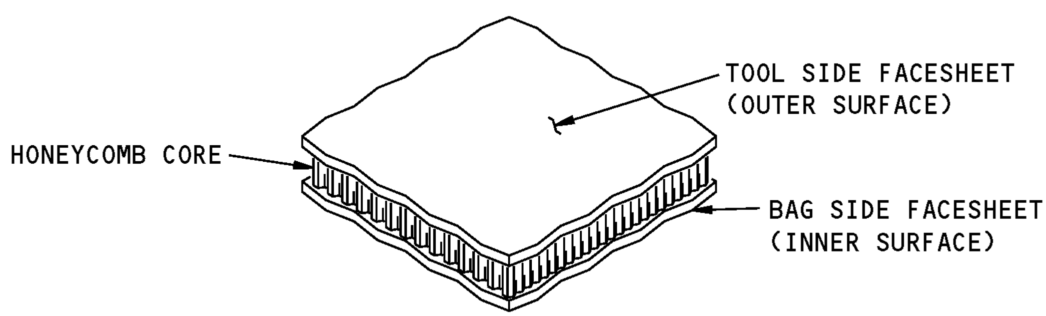
F. Install the wing fixed trailing edge panels that were removed.

(1) Install the fasteners after all repairs are complete. Refer to 51-40-02 for information on fastener installation.

G. Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.




**Definitions of the Damage Size**
**Figure 202**




**Definitions of the Facesheets**
**Figure 203**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-05 | REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS |
| 51-70-06, REPAIR GENERAL | Room Temperature Cure Repairs With Wet Layup Materials For Glass Fabric Reinforced Plastic Solid Laminates and Honeycomb Core Panels |
| 51-70-13 | TYPICAL WEB REPAIRS |
| 51-70-14 | STRUCTURES WITH ALUMINUM COATINGS AND FOILS |
| 737 NDT Part 1, 51-01-01 | Inspection of Repairs to Composite Structure |

## 4. Repair Instructions for the 250 Degree F (121 Degree C) Cure Panels

A. For dents that are a maximum of 2 inches in diameter and have no fiber damage or delamination, do the steps that follow:

(1) Fill the dent with BMS 5-28, Type 7 potting compound

(2) Apply a fiberglass patch over the potted area as given in 51-70-04.

B. For dents that are not permitted by Paragraph 4.A./REPAIR 1 and for other damage that is not permitted by Allowable Damage 1, refer to:

(1) Table 201/REPAIR 1 for panel areas other than the edgebands

(2) Table 202/REPAIR 1 for the edgebands.

C. For repairs with wet layup materials, do as follows:

(1) Use the same number of repair plies as the number of initial plies that were removed.

(2) Add two structural plies for each facesheet that is repaired. Put one ply at $\pm 45$ and the other ply at 0 or 90 degrees.

(3) Inspect Category B repairs after each 800 flight hour interval or more frequently.

(a) Refer to 737 NDT Part 1, 51-01-01 for the inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(b) Replace the repair with a Category A repair if you find deterioration.

D. Use the instructions that follow to do a Category A repair with preimpregnated layup materials at 250°F (121°C) cure.

(1) Repair the damage as given in 51-70-05.

(2) Use the same number of repair plies as the number of initial plies that were removed.

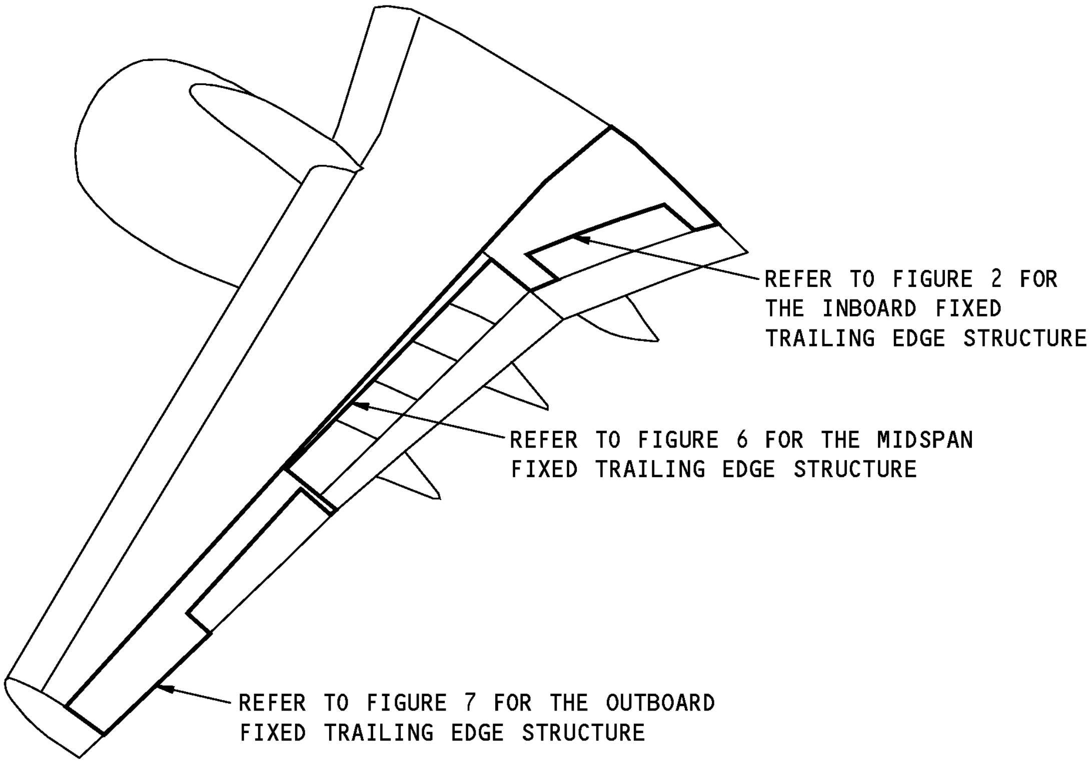
Table 201:

| REPAIR DATA FOR THE 250°F (121°C) CURE TRAILING EDGE SKIN PANELS FOR AREAS OTHER THAN THE EDGEBANDS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **Room Temperature** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** |
| REPAIR SIZE | Damage that is a maximum of:<br><br>- 2.0 inches in diameter<br>- 30 percent of the smallest dimension across the panel at the damage location<br>- One facesheet and the honeycomb core in depth | Damage that is a maximum of:<br><br>- 4.0 inches in diameter<br>- 50 percent of the smallest dimension across the panel at the damage location | Damage that is a maximum of:<br><br>- 6.0 inches in diameter<br>- 50 percent of the smallest dimension across the panel at the damage location | There are no limits on the dimensions of the repair |
| | One repair for each 144 square inches | One repair for each 144 square inches | One repair for each 144 square inches | |
| | 6.0 inches minimum clearance from:<br><br>- other repairs (includes repairs on the opposite panel surface)<br>- fastener holes<br>- damage permitted in Allowable Damage 1<br>- panel edges | 4.0 inches minimum clearance from:<br><br>- other repairs (includes repairs on the opposite panel surface)<br>- fastener holes<br>- damage permitted in Allowable Damage 1<br>- panel edges | 4.0 inches minimum clearance from:<br><br>- other repairs (includes repairs on the opposite panel surface)<br>- fastener holes<br>- damage permitted in Allowable Damage 1<br>- panel edges | |
| REPAIR INSTRUCTIONS | SRM 51-70-06 and Paragraph 4.C | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-05 and Paragraph 4.D |

Table 202:

| REPAIR DATA FOR THE EDGEBANDS OF 250°F (121°C) CURE TRAILING EDGE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **Room Temperature** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** |
| REPAIR SIZE | Damage that is a maximum of:<br><br>- 15 percent of the cross-sectional area of the edgeband at the damage location | Damage that is a maximum of:<br><br>- 15 percent of the cross-sectional area of the edgeband at the damage location | There are no limits on the dimensions of the repair | There are no limits on the dimensions of the repair |
| | - 10 percent of the length of the edgeband on the side of the damage | - 10 percent of the length of the edgeband on the side of the damage | | |
| REPAIR INSTRUCTIONS | SRM 51-70-06 and Paragraph 4.C | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-05 and Paragraph 4.D |

## IDENTIFICATION 1 - WING FIXED TRAILING EDGE STRUCTURE



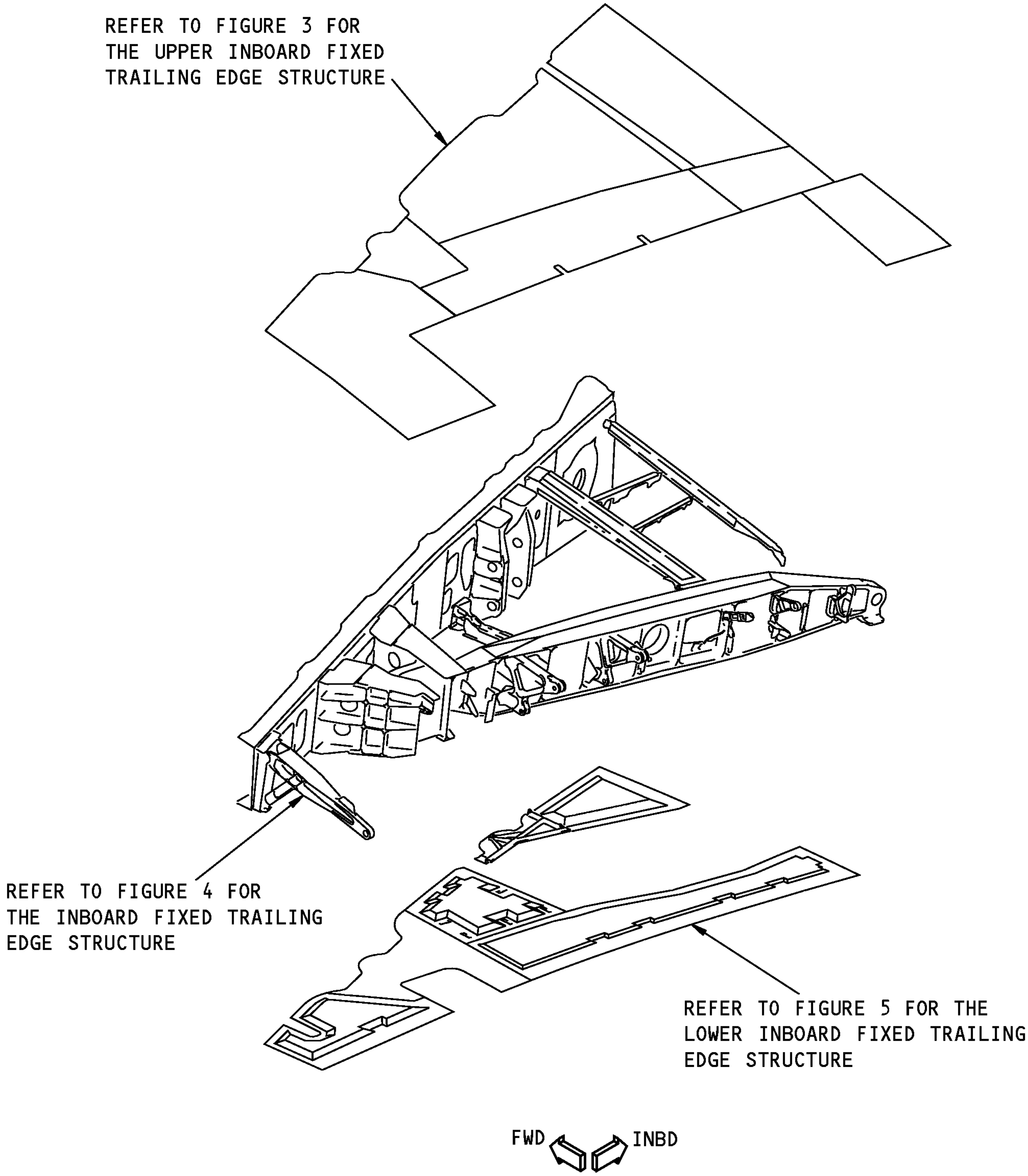

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Fixed Trailing Edge Location**
**Figure 1**

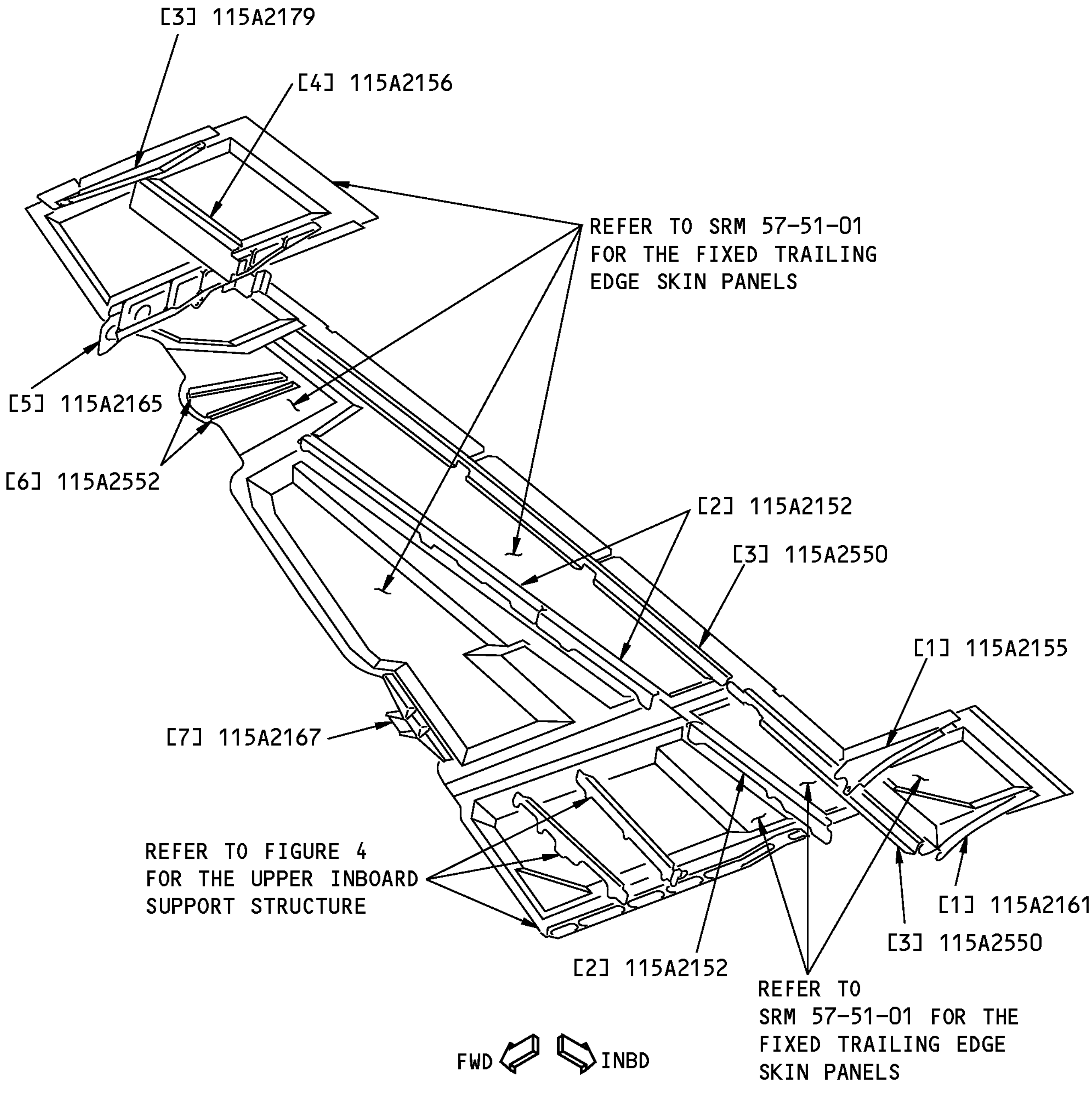
**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 115A0001 | Functional Product Collector, Fixed Trailing Edge - Left Wing |
| 115A0002 | Functional Product Collector, Fixed Trailing Edge - Right Wing |
| 115A0003 | Functional Product Collector, Fixed Trailing Edge to Rear Spar - Left Wing |
| 115A0004 | Functional Product Collector, Fixed Trailing Edge to Rear Spar - Right Wing |
| 115A2000 | Upper Installation - Inboard Fixed Trailing Edge |
| 115A2001 | Lower Installation - Inboard Fixed Trailing Edge |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 115A2500 | Panel Installation - Inboard Fixed Trailing Edge |
| 115A3700 | Panel Installation - Midspan Fixed Trailing Edge, Lower |
| 115A4900 | Structure Installation - Panels and Supports, Outboard Fixed Trailing Edge |
| 115A6200 | Rib Installation - Deflection Control, Midspan Fixed Trailing Edge |
| 115A8100 | Rib Installation - WBL 413.18 and 419.75, Outboard Fixed Trailing Edge |
| 115A8300 | Support Installation - Aileron Hinge, Outboard Fixed Trailing Edge |



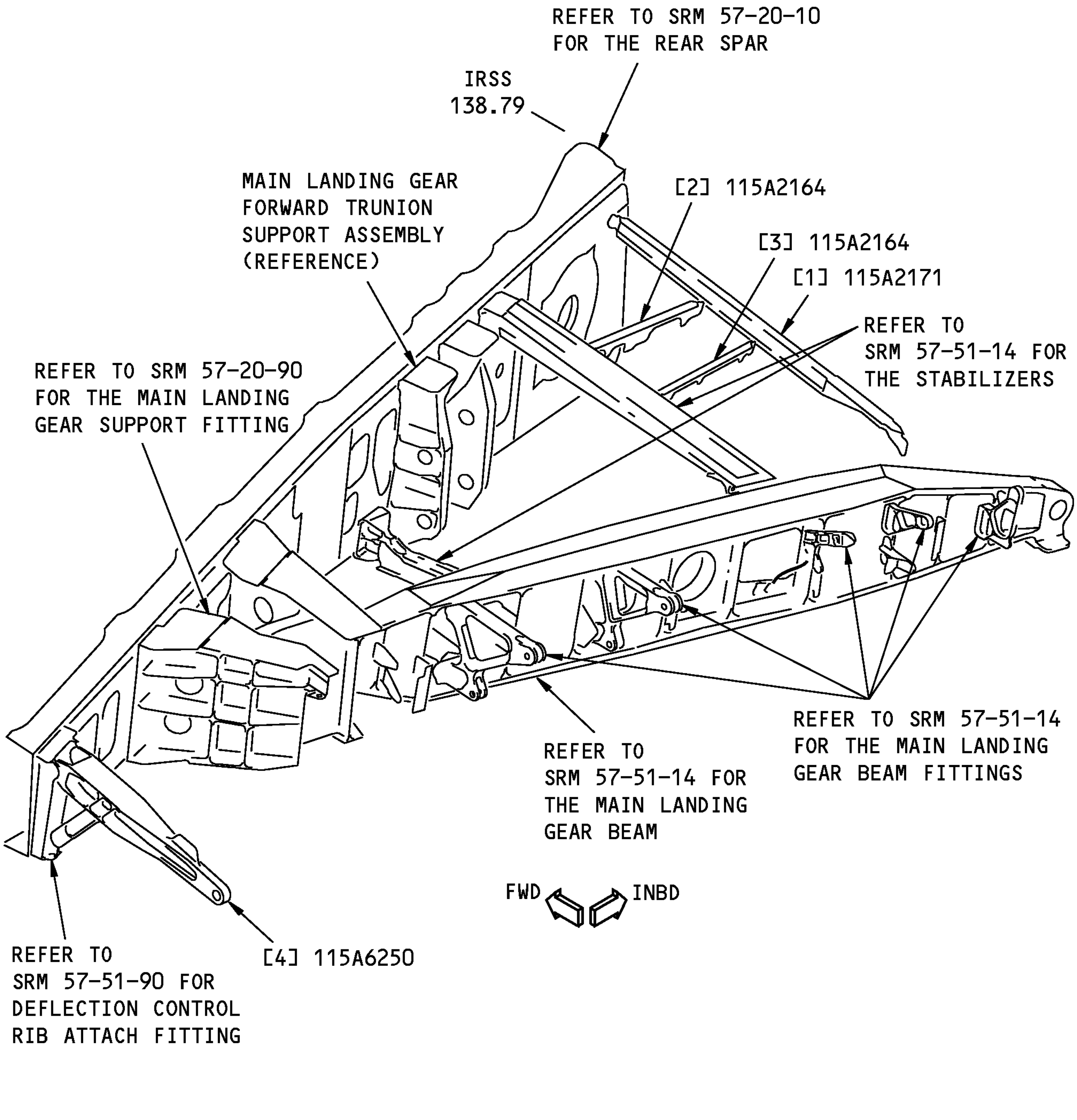

**Inboard Fixed Trailing Edge Structure**
**Figure 2**




Upper Inboard Fixed Trailing Edge Structure
Figure 3

| LIST OF MATERIALS FOR FIGURE 3 | | | |
|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL |
| [1] | Panel Support Bracket (2) | | 7050-7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [2] | Panel Support J Beam (2) | | BAC1505-100769 extrusion as given in QQ-A-200/11. |
| [3] | Panel Support Beam | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [4] | Panel Support Angle | 0.050 (1.27) | 2024-T3 bare sheet as given in QQ-A-250/4. |
| [5] | Panel Support Fitting | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [6] | Panel Stiffener (2) | | AND10134-1004 7075-T62 extrusion as given in QQ-A-200/11. |
| [7] | Panel Support Bracket | | 7075-T7351 machined plate as given in QQ-A-250/12. Grain direction controlled part. |

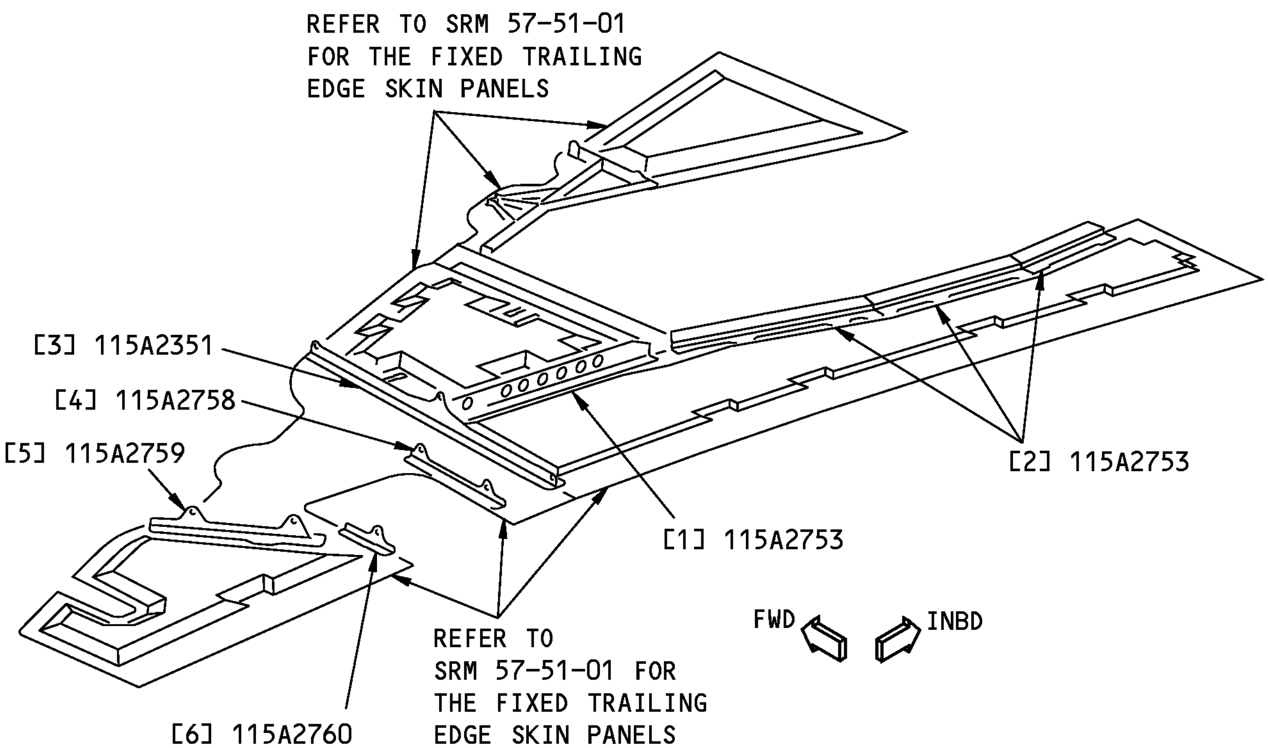
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Inboard Fixed Trailing Edge Structure**
**Figure 4**

| LIST OF MATERIALS FOR FIGURE 4 | | | |
|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL |
| [1] | Support Beam | | Ti-6Al-4V machined plate as given in Mil-T-9046, Code AB-1, in the annealed condition. |
| [2] | Systems Tray | | BAC1505-101134 7075-T6511 extrusion as given in QQ-A-200/11. |
| [3] | Systems Tray | | BAC1503-100093 7075-T6511 extrusion as given in QQ-A-200/11. |
| [4] | Deflection Control Rib | | 7050-T7451 machined plate as given in AMS 4050. |

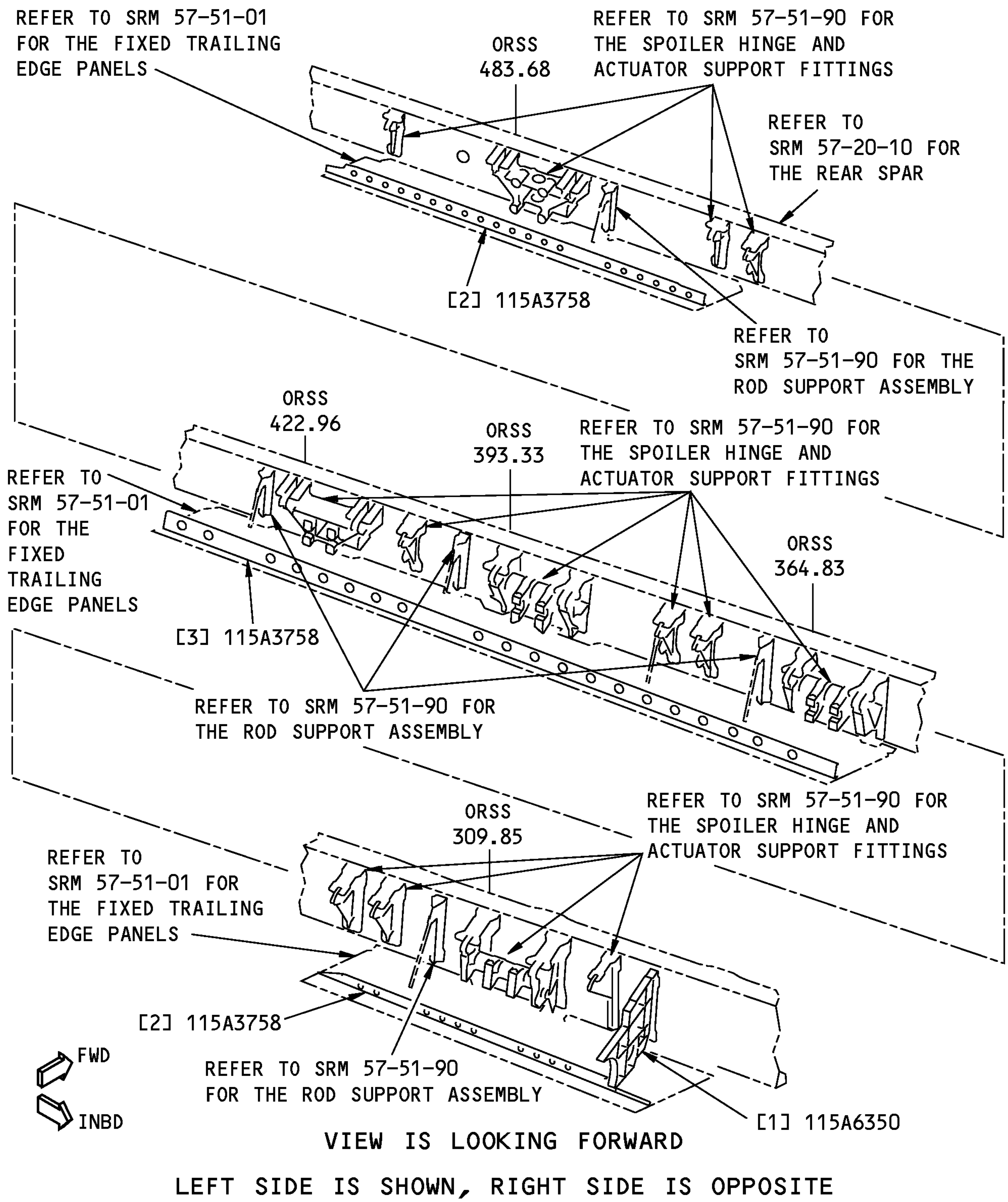
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Lower Inboard Fixed Trailing Edge Structure**
**Figure 5**

| LIST OF MATERIALS FOR FIGURE 5 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Panel Support Beam | | BAC1505-101301 7075-T73511 extrusion as given in QQ-A-200/11. | |
| [2] | Panel Support Beam (3) | | BAC1510-1335 7075-T73511 extrusion as given in QQ-A-200/11. | |
| [3] | Panel Support Beam | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. | |
| [4] | Panel Support | | BAC1514–918 7075-T73511 extrusion as given in QQ-A-200/11. | |
| [5] | Panel Support Beam | | BAC1506-3876 7075-T73511 extrusion as given in QQ-A-200/11. Optional: 7050-T7451 machined plate as given in BMS 7-323. | LINE NUMBER 1 TO 1980 |
| [5] | Panel Support Beam | | 7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part. | LINE NUMBER 1981 AND ON |
| [6] | Panel Support | | 7075-T7351 machined plate as given in QQ-A-250/12. Grain direction controlled part. | |

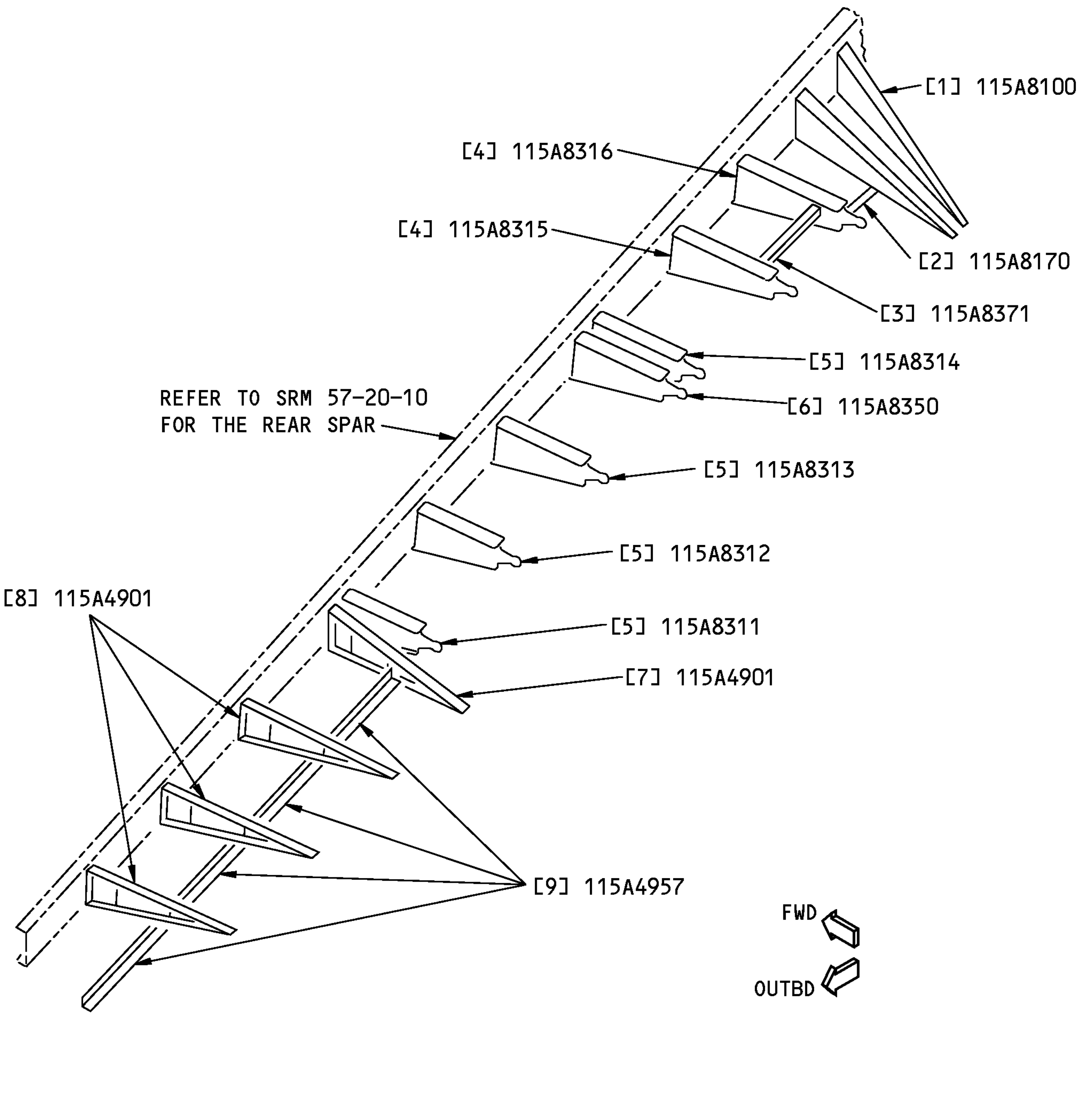
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Midspan Fixed Trailing Edge Structure**
**Figure 6**

| LIST OF MATERIALS FOR FIGURE 6 | | | |
|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL |
| [1] | Seal Bulkhead | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [2] | Lower Panel Support Beam (2) | | BAC1506–1489 7075-T73511 extrusion as given in QQ-A-200/11. |
| [3] | Lower Panel Support Beam | | BAC1509–100590 7075-T73511 extrusion as given in QQ-A-200/11. |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Outboard Fixed Trailing Edge Structure**
**Figure 7**

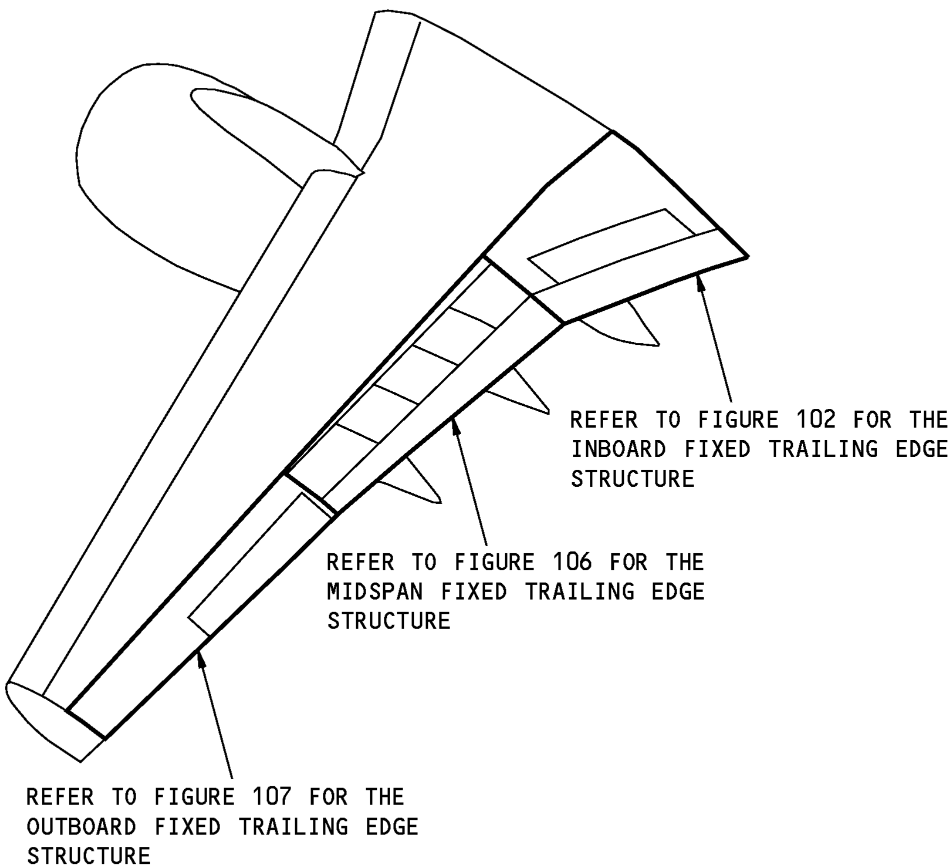
| LIST OF MATERIALS FOR FIGURE 7 | | | |
|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL |
| [1] | Rib Assembly | | |
| | Rib (2) | | 7050-T7451 machined plate as given in AMS 4050. |
| | Angle | 0.040 (1.0) | 2024-T3 clad sheet as given in QQ-A-250/5. |
| | Intercostal | 0.040 (1.0) | 2024-T3 clad sheet as given in QQ-A-250/5. |
| [2] | Lower Panel Support Beam | | 7050-T7451 machined plate as given in AMS 4050. |
| [3] | Panel Support Angle | | 7050-T7451 machined plate as given in AMS 4050. |
| [4] | Aileron Hinge Support Assembly | | |
| | Rib | | 7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part. |
| | Bearing Link | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [5] | Aileron Hinge Support Assembly | | |
| | Rib | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| | Bearing Link | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [6] | Aileron Tab Control Rod Support Assembly | | |
| | Rib | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| | Clevis | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [7] | Panel Support Rib | | |
| | Upper Chord | | 7075-T7451 machined plate as given in QQ-A-250/12. Grain direction controlled part. |
| | Lower Chord | | 7075-T7451 machined plate as given in QQ-A-250/12. Grain direction controlled part. |
| | Stiffener (2) | 0.040 (1.0) | 2024-T3 clad sheet as given in QQ-A-250/5. |
| | Web | 0.040 (1.0) | 2024-T3 clad sheet as given in QQ-A-250/5. |
| [8] | Panel Support Rib | | |
| | Upper Chord | | 7075-T7451 machined plate as given in QQ-A-250/12. Grain direction controlled part. |
| | Lower Chord | | AND10139-2405 7075-T73511 extrusion as given in QQ-A-200/11. |
| | Stiffener (2) | 0.040 (1.0) | 2024-T3 clad sheet as given in QQ-A-250/5. |
| | Web | 0.040 (1.0) | 2024-T3 clad sheet as given in QQ-A-250/5. |
| [9] | Panel Support Beam | | BAC1505-29542 7075-T73 extrusion as given in QQ-A-200/11. |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

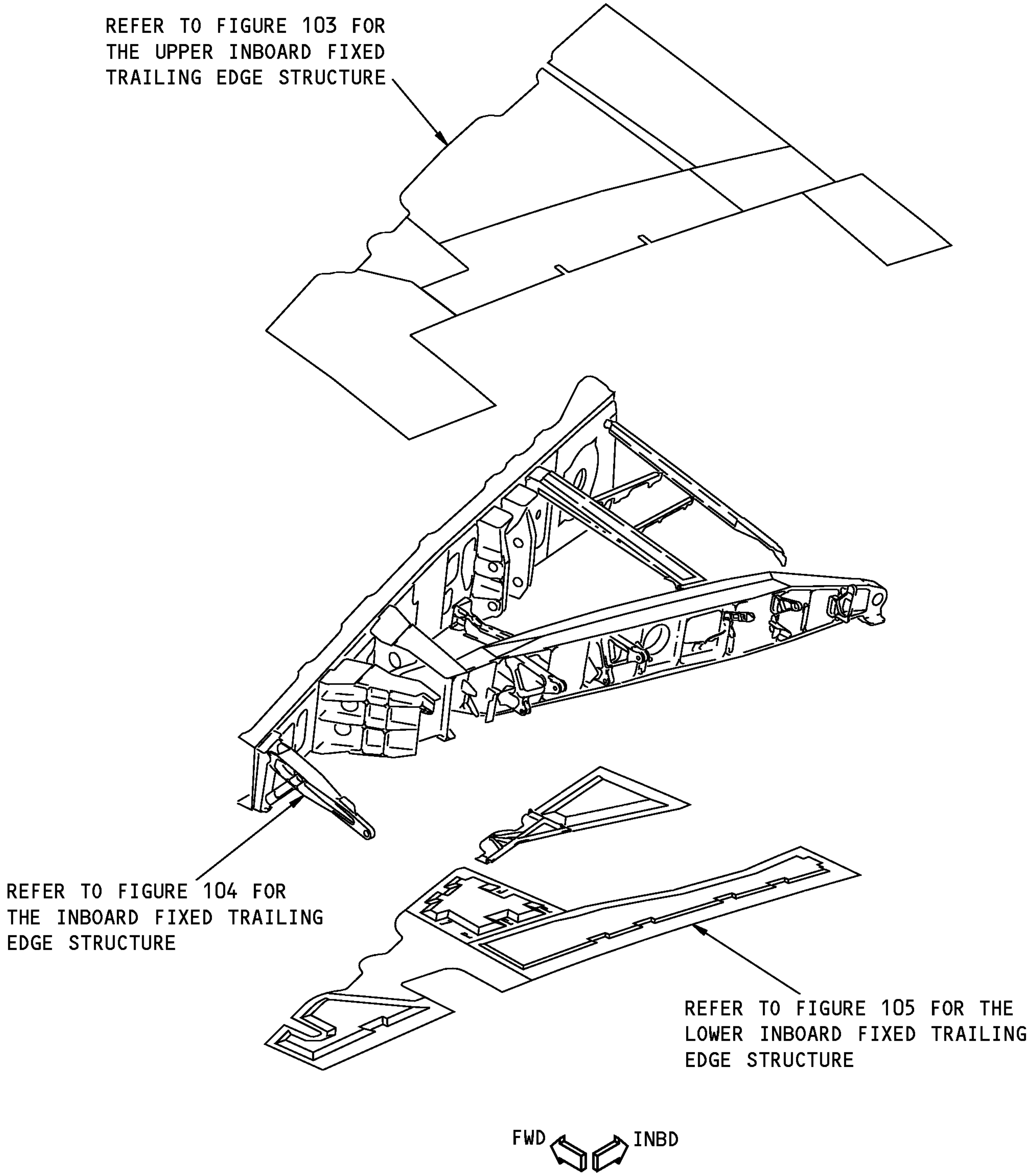
## ALLOWABLE DAMAGE 1 - WING FIXED TRAILING EDGE STRUCTURE

### 1. Applicability

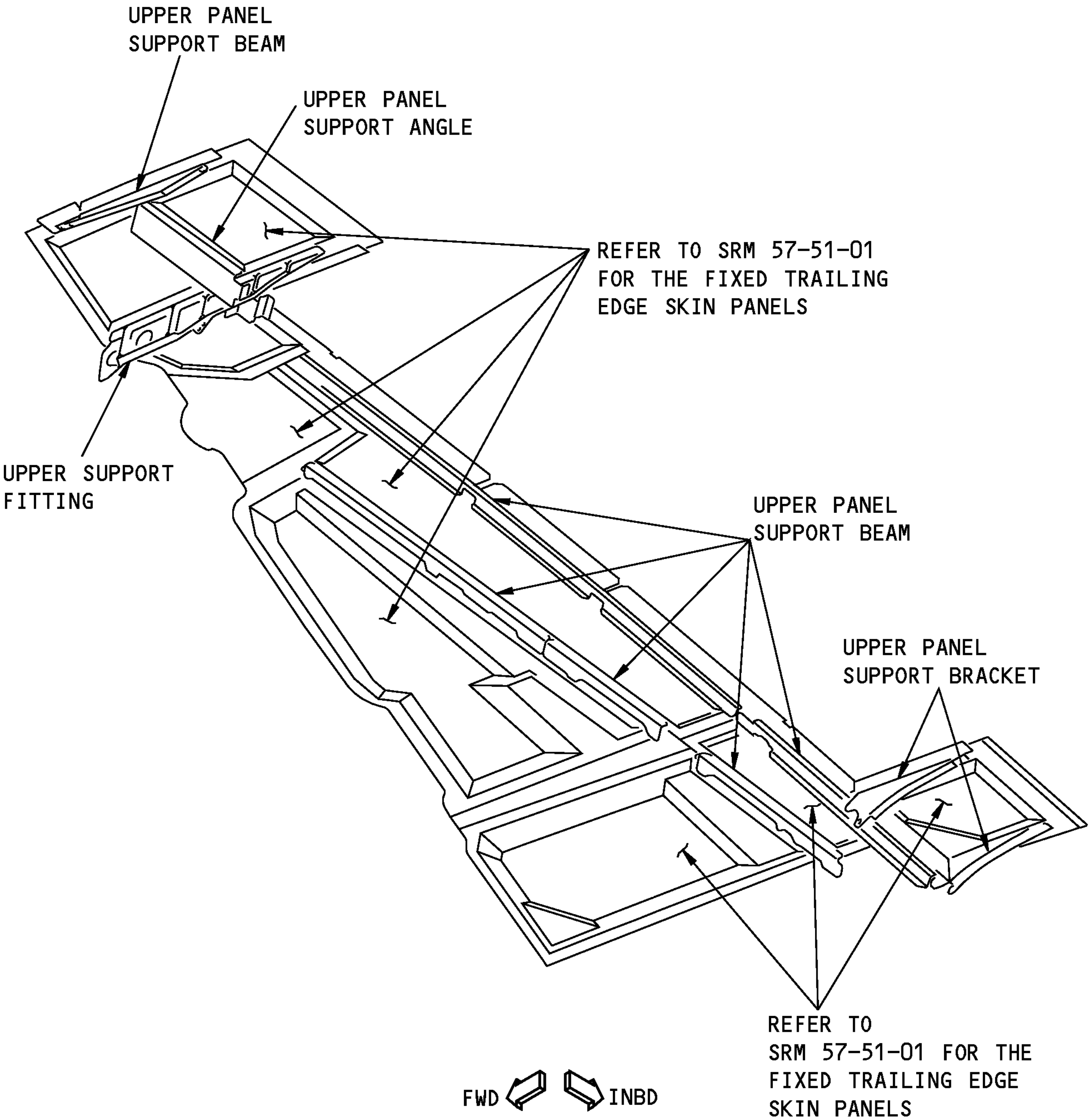
A. Allowable Damage 1 is applicable to damage on the wing fixed trailing edge structure shown in Wing Fixed Trailing Edge Location, Figure 101/ALLOWABLE DAMAGE 1.



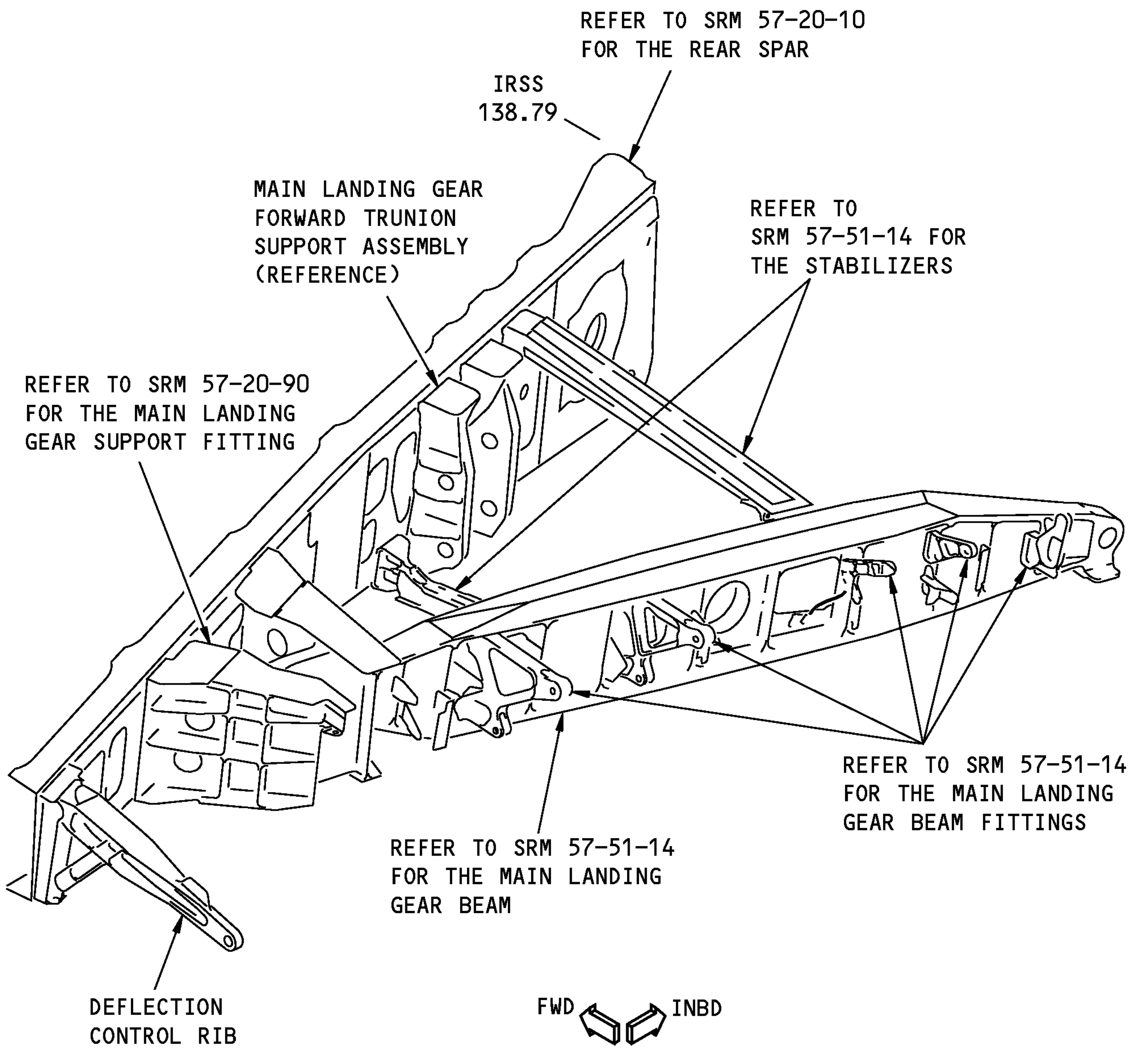

**Wing Fixed Trailing Edge Location**
**Figure 101**



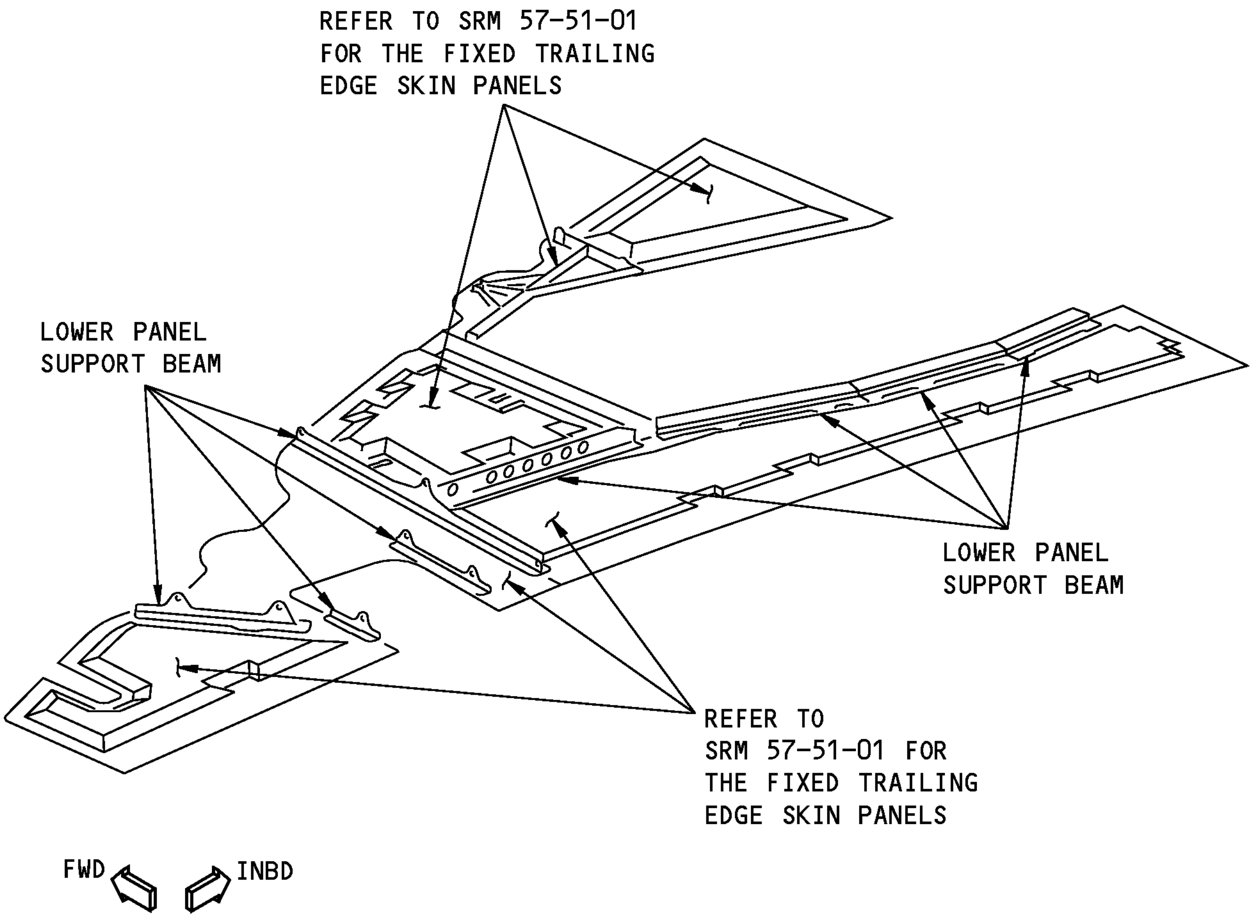

**Inboard Fixed Trailing Edge Structure**
**Figure 102**



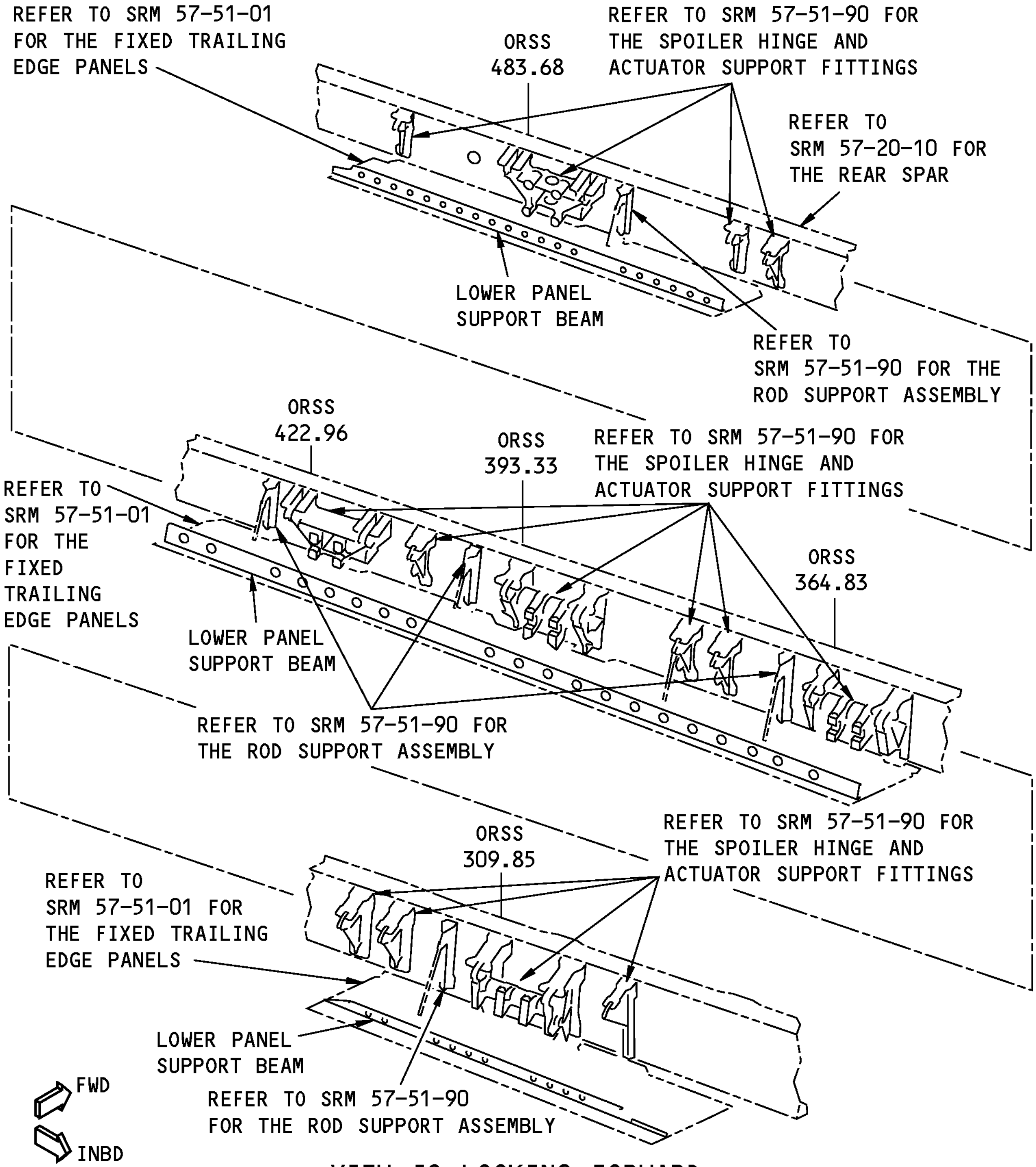

**Upper Inboard Fixed Trailing Edge Structure**
**Figure 103**




**Inboard Fixed Trailing Edge Structure**
**Figure 104**



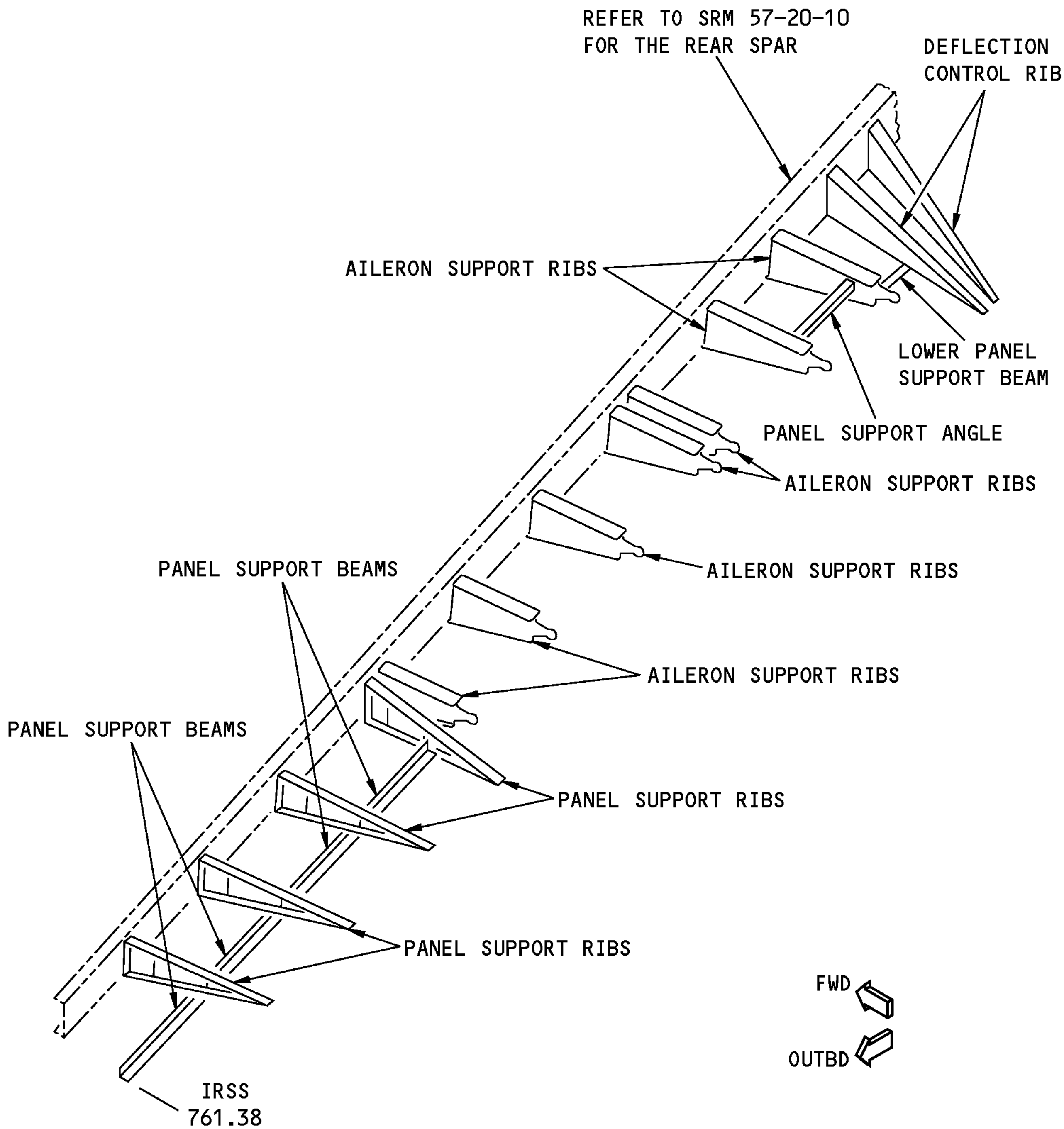

**Lower Inboard Fixed Trailing Edge Structure**
**Figure 105**




VIEW IS LOOKING FORWARD

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Midspan Wing Fixed Trailing Edge Structure**
**Figure 106**




**Outboard Wing Fixed Trailing Edge Structure**
**Figure 107**

## 2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limit references that are applicable to each type of structure.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| TYPE OF STRUCTURE | PARAGRAPH |
| PANEL SUPPORT RIBS | 4.A |
| PANEL SUPPORT STRUCTURE (DOES NOT INCLUDE PANEL SUPPORT RIBS) | 4.B |
| AILERON SUPPORT RIBS | 4.C |
| DEFLECTION CONTROL RIBS | 4.D |

C. Remove the damage.

(1) Refer to 51-10-02, GENERAL for the investigation and cleanup procedures.

(2) Refer to 51-30-03, GENERAL for possible sources of the abrasive and other materials you can use to remove the damage.

(3) Refer to 51-30-05, GENERAL for possible sources of the equipment and tools you can use to remove the damage.

D. After you remove the damage, do as follows:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the parts given in Table 102/ALLOWABLE DAMAGE 1.

(a) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

**Table 102**:

| PARTS THAT YOU MUST SHOT PEEN OR FLAP PEEN AFTER YOU REMOVE THE DAMAGE | |
|---|---|
| PART DRAWING NUMBER | PART NAME |
| 115A6250 | Rib Assembly - Deflection Control - Midspan |
| 115A6252 | Clevis Fitting Assembly - Deflection Control - Midspan |

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01, GENERAL.

(3) Apply two layers of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-44-04.

## 3. References

| Reference | Title |
|---|---|
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01, GENERAL | Protective Treatment of Metallic and Composite Materials |
| 51-20-06, GENERAL | Shot Peening |

(Continued)

| Reference | Title |
|---|---|
| 51-30-03, GENERAL | Sources for Non-Metallic Repair Materials |
| 51-30-05, GENERAL | Equipment and Tools For Repairs |
| 51-40-05, GENERAL | Fastener Hole Sizes |
| 57-51-02, REPAIR 1 | Wing Fixed Trailing Edge Support Beam |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Panel Support Ribs

(1) Rib Chords

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Rib Webs

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A and E .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , C , D , and E.

2) Damage that does not go though the clad surface is permitted.

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if they are:

1) A minimum of 1.0 inch away from the edge of a fastener, a hole, other damage, or material edge.

2) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

a) Use a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

b) Drill the holes to the applicable size as given in 51-40-05, GENERAL.

(3) Rib Stiffeners

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

2) Damage that does not go through the clad surface is permitted.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

B. Panel Support Structure (Does not include Panel Support Ribs)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Aileron Support Ribs

(1) Rib Chord Areas

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Rib Web Areas

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail C.

(c) Dents are permitted as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail F.

(d) Holes and Punctures are permitted if they are:

1) A minimum of 1.0 inch away from the edge of a fastener, a hole, other damage, or material edge

2) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

a) Use a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

b) Drill the holes to the applicable size as given in 51-40-05, GENERAL.

D. Deflection Control Ribs

(1) Rib Chord Areas

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Rib Web Areas

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail C.

(c) Dents are permitted as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail F.

(d) Holes and Punctures are permitted if they are:

1) A minimum of 1.0 inch away from the edge of a fastener hole, other damage, or material edge

2) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

a) Use an aluminum rivet installed without sealant.

b) Drill the holes to the applicable size as given in 51-40-05, GENERAL.

(3) Rib Upper Flange

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

**NOTE**: If you have done 57-51-02, REPAIR 1, then Paragraph 4.D.(3)(b)1)/ALLOWABLE DAMAGE 1 does not apply. Use Paragraph 4.D.(3)(b)2)/ALLOWABLE DAMAGE 1

1) The damage is permitted as given in Figure 108/ALLOWABLE DAMAGE 1, Detail G if:

a) You remove the initial BACB30NM3K8 bolt or BACB30LE3DK bolt, as applicable.

b) You do not blend more than 0.04 inch (1.02 mm) thickness from the upper vertical flange.

c) You install either a BACW10P440CG or BACW10P461CG washer as shown in Figure 203/57-51-02, REPAIR 1, Detail G.

d) You install a new BACB30LE3DK8 bolt as givenn in drawing 115A6200.

2) If you have done 57-51-02, REPAIR 1, the damage is permitted as given in Figure 108/ALLOWABLE DAMAGE 1, Detail G if:

a) You remove the initial BACB30US4K( )DM bolts.

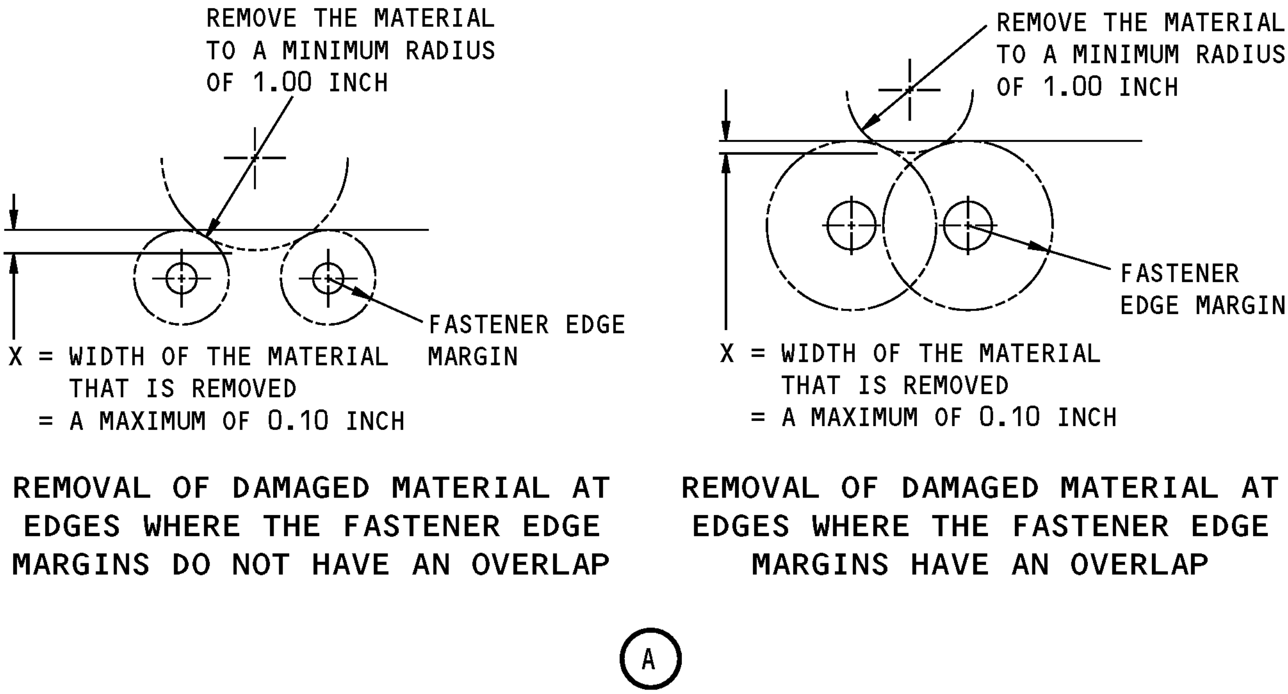
b) You do not blend more than 0.04 inch (1.02 mm) thickness from the upper vertical flange.

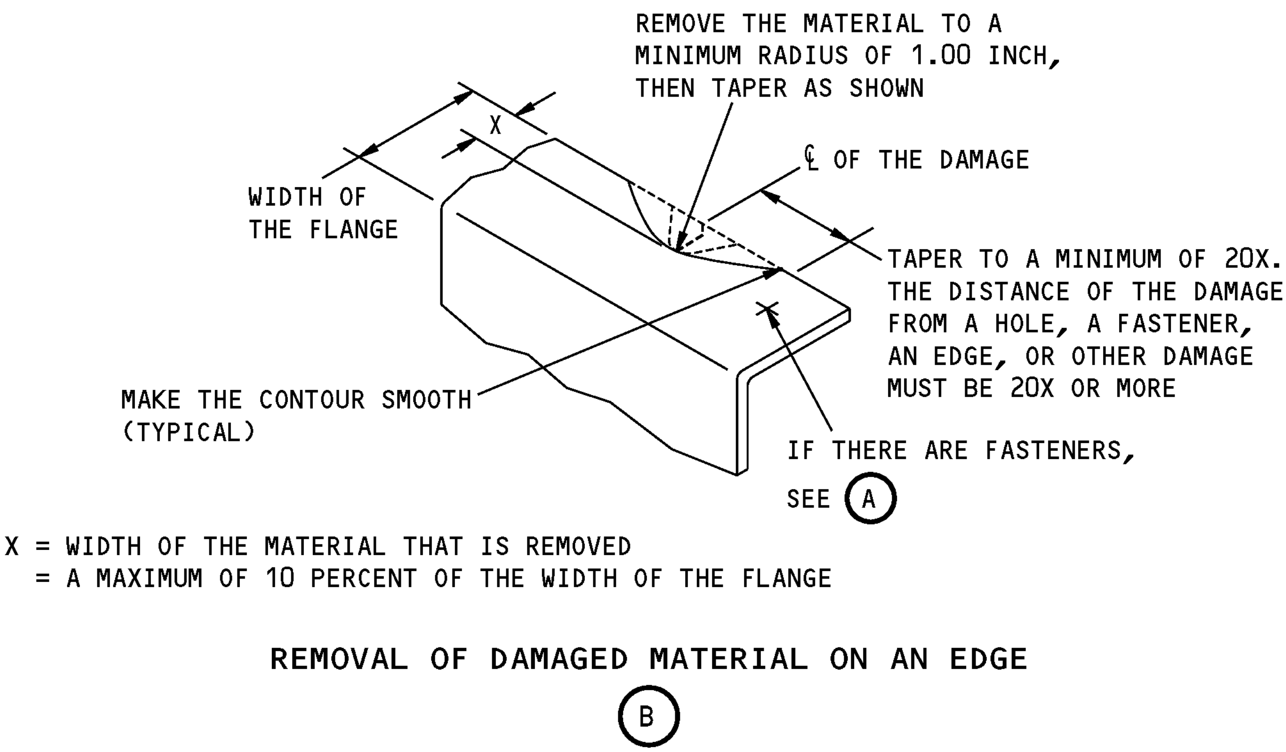
c) You re-install the BACW10P453CG and the BACW10BP4AP washers as shown in Figure 203/57-51-02, REPAIR 1.

d) You re-install the BACB30US4K( )DM bolts as shown in Figure 203/57-51-02, REPAIR 1.

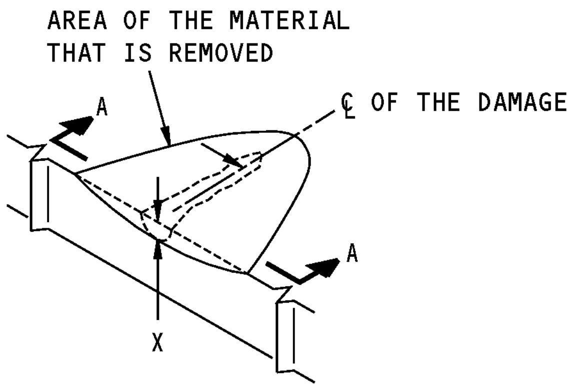
(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.



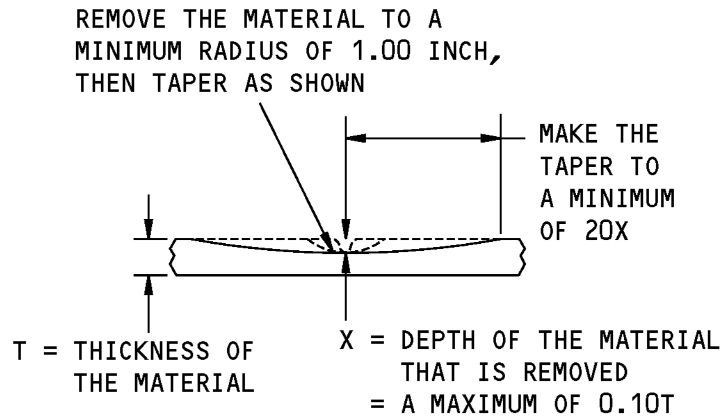




**Allowable Damage Limits**
**Figure 108 (Sheet 1 of 4)**

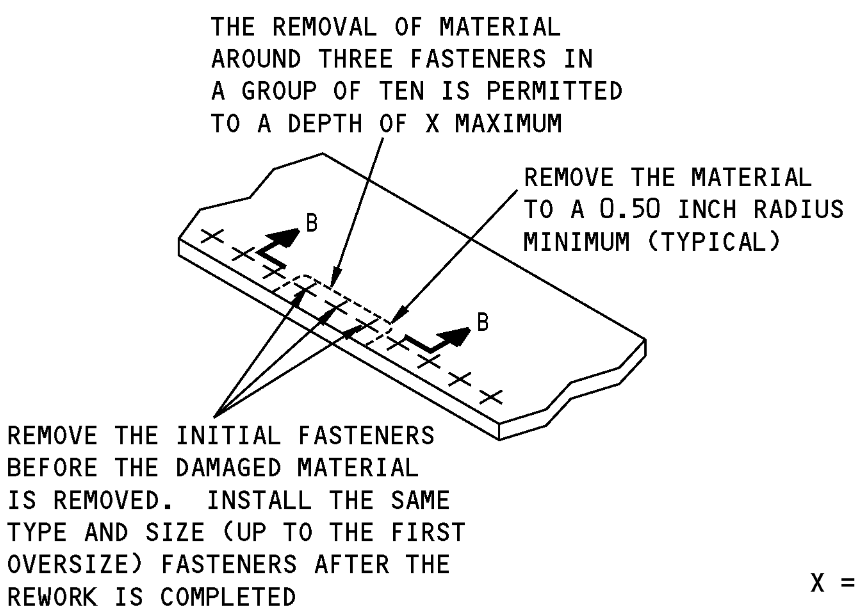



REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

C

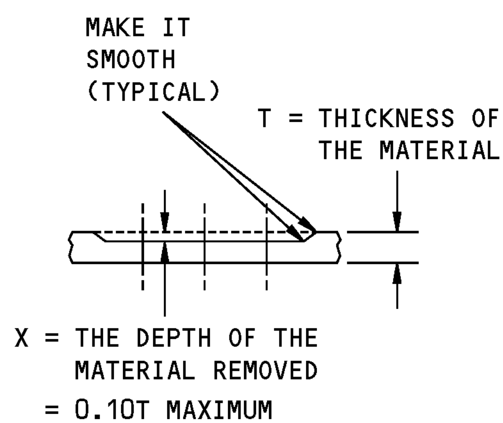



A-A



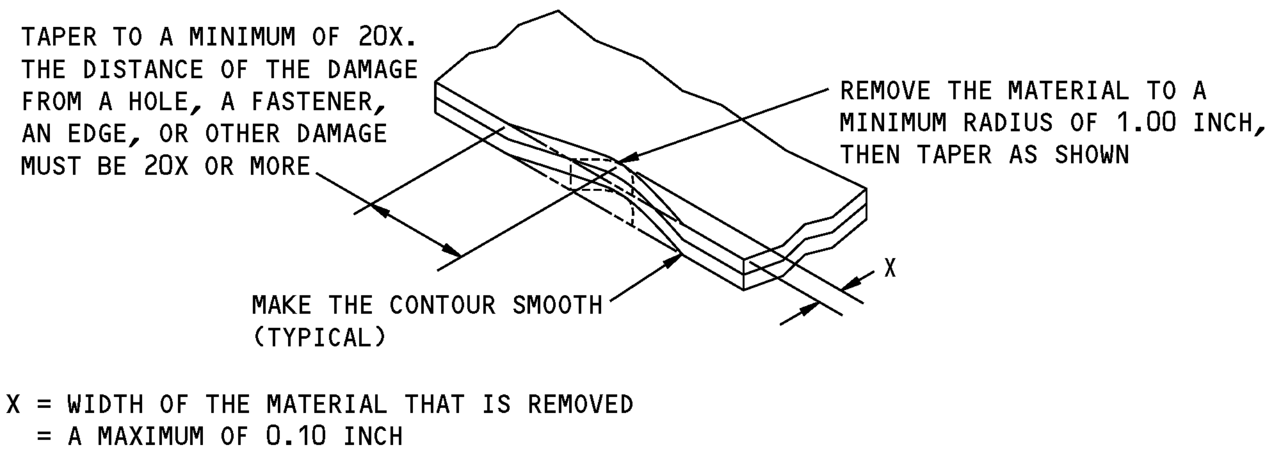

REMOVAL OF CORROSION
AROUND THE FASTENERS

D



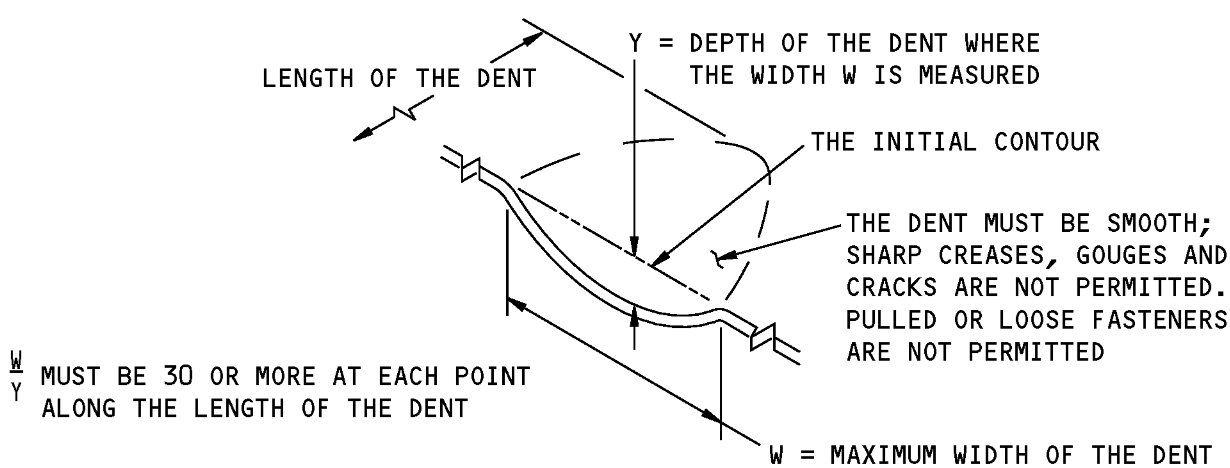

B-B

**Allowable Damage Limits**
**Figure 108 (Sheet 2 of 4)**




REMOVAL OF DAMAGED MATERIAL
ON AN EDGE OF A WEB

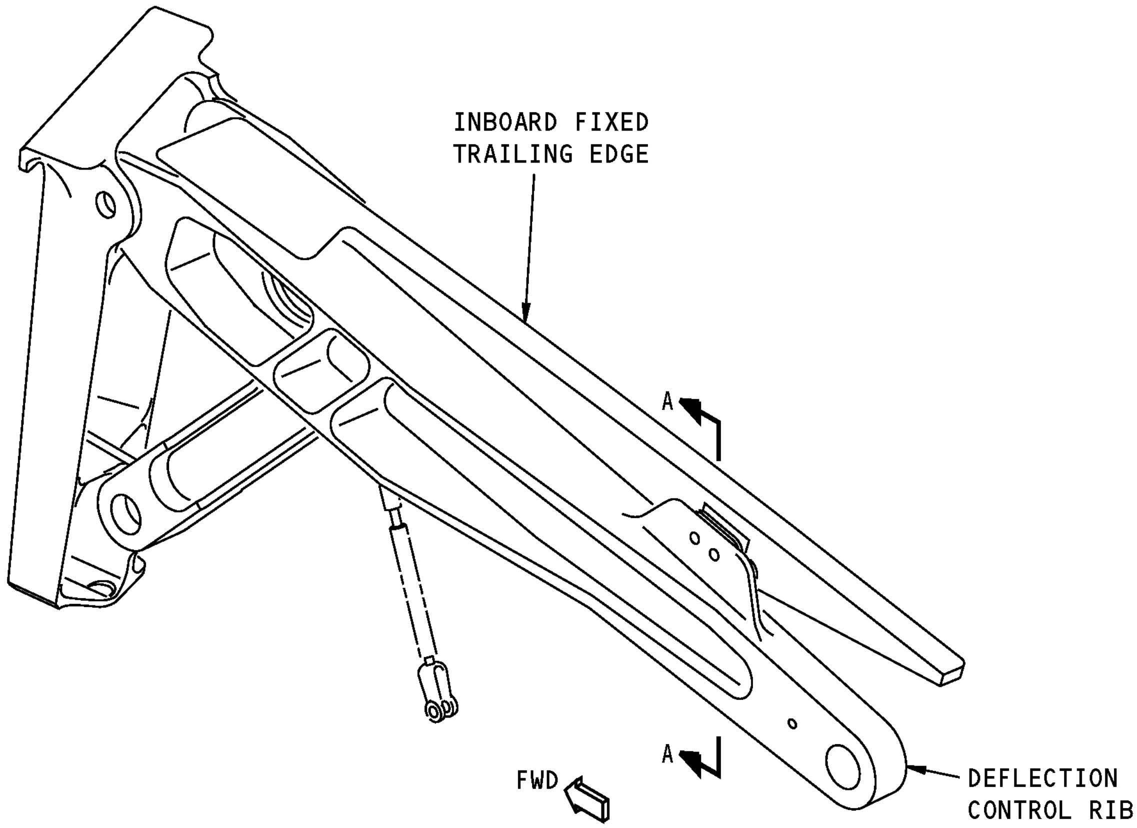
E




DENT THAT IS PERMITTED

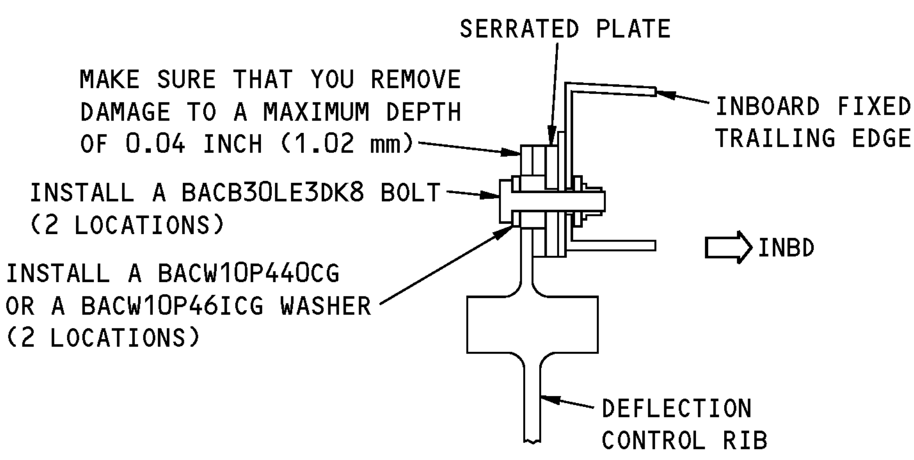
F

**Allowable Damage Limits**
**Figure 108 (Sheet 3 of 4)**




UPPER FLANGE DAMAGE THAT IS PERMITTED

G



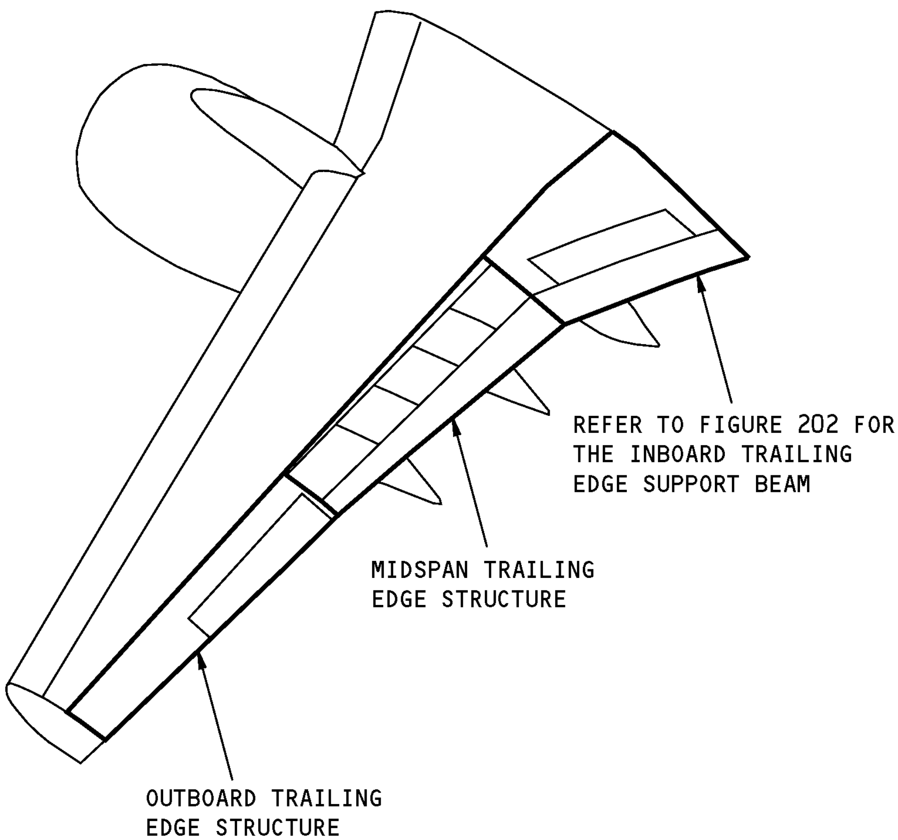

A-A

**Allowable Damage Limits**
**Figure 108 (Sheet 4 of 4)**

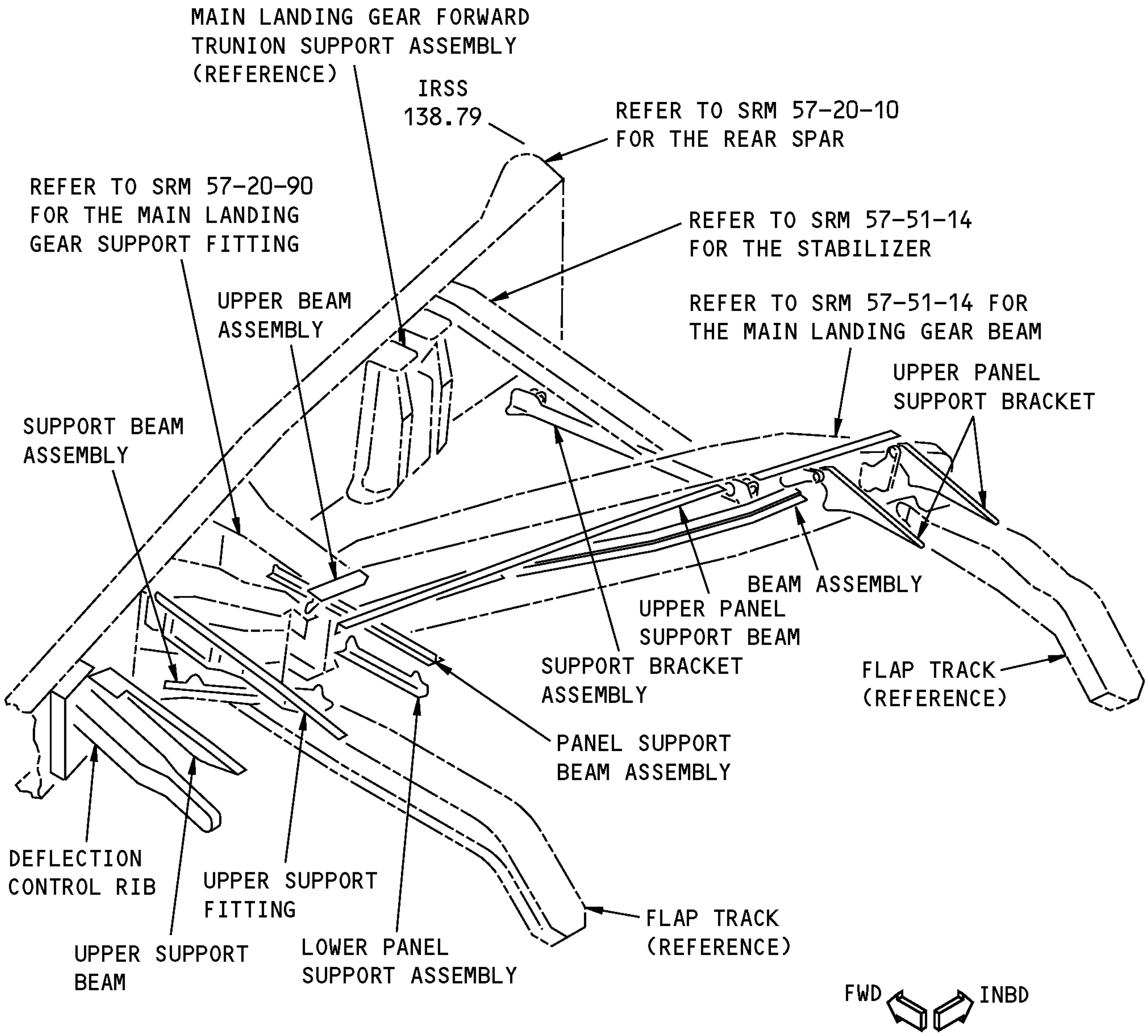
## REPAIR 1 - WING FIXED TRAILING EDGE SUPPORT BEAM

### 1. Applicability

A. Repair 1 is applicable to damage to the wing fixed trailing edge support beam as shown in Figure 201.




**Wing Fixed Trailing Edge Location**
**Figure 201**




NOTES

- NOT ALL OF THE FIXED TRAILING EDGE STRUCTURE IS SHOWN.

**Wing Outboard Trailing Edge Structure**
**Figure 202**

## 2. General

A. Repair 1 is a permanent repair. Refer to 51-00-06, GENERAL to find the definitions of the different categories of repairs.

## 3. References

| Reference | Title |
|---|---|
| 51-00-06, GENERAL | Structural Repair Definitions |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01, GENERAL | Protective Treatment of Metallic and Composite Materials |
| 51-20-05, GENERAL | Repair Sealing |
| 51-40-02, GENERAL | Fastener Installation and Removal |
| 51-40-03, GENERAL | Fastener Substitution |
| 57-51-02, ALLOWABLE DAMAGE 1 | Wing Fixed Trailing Edge Structure |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-42-05 | Bright Cadmium Plating |

## 4. Repair Instructions

A. Trim out and remove all damage in the trailing edge upper support beam as shown in Figure 203 to a maximum slot of 1.20 inch (30.48 mm) by 0.56 inch (14.22 mm). Make sure that you use a minimum trimmout radius of 0.11 inch (2.79 mm).

(1) Make sure that you can install the 0.25 inch (6.35 mm) diameter BACB30US4K( )DM bolt through all slots and holes as given in Paragraph 4.J.(1)/REPAIR 1. If necessary, increase the dimension of the slots or holes to 0.250/0.277 inch (6.35/7.04 mm).

B. Make sure that all of the damage is removed. Refer to 51-10-02, GENERAL for the removal of damage procedures, and to 57-51-02, ALLOWABLE DAMAGE 1 for damage to the deflection control rib.

C. Make the part [1] repair doubler as shown in Figure 203. Refer to Table 201 for the repair material.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Doubler | 1 | Use 15-5PH CRES as given in AMS 5659 with a heat treat of 180 to 200KSI and that is a minimum of 0.30 inch (7.62 mm) thick. |

D. Assemble the part [1] repair doubler and drill the fastener holes as shown in Figure 203.

E. Disassemble the part [1] repair doubler.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair part and to the bare surfaces of the initial structure.

G. Restore the initial structure finish as given in 51-20-01, GENERAL.

H. Finish the part [1] repair doubler with:

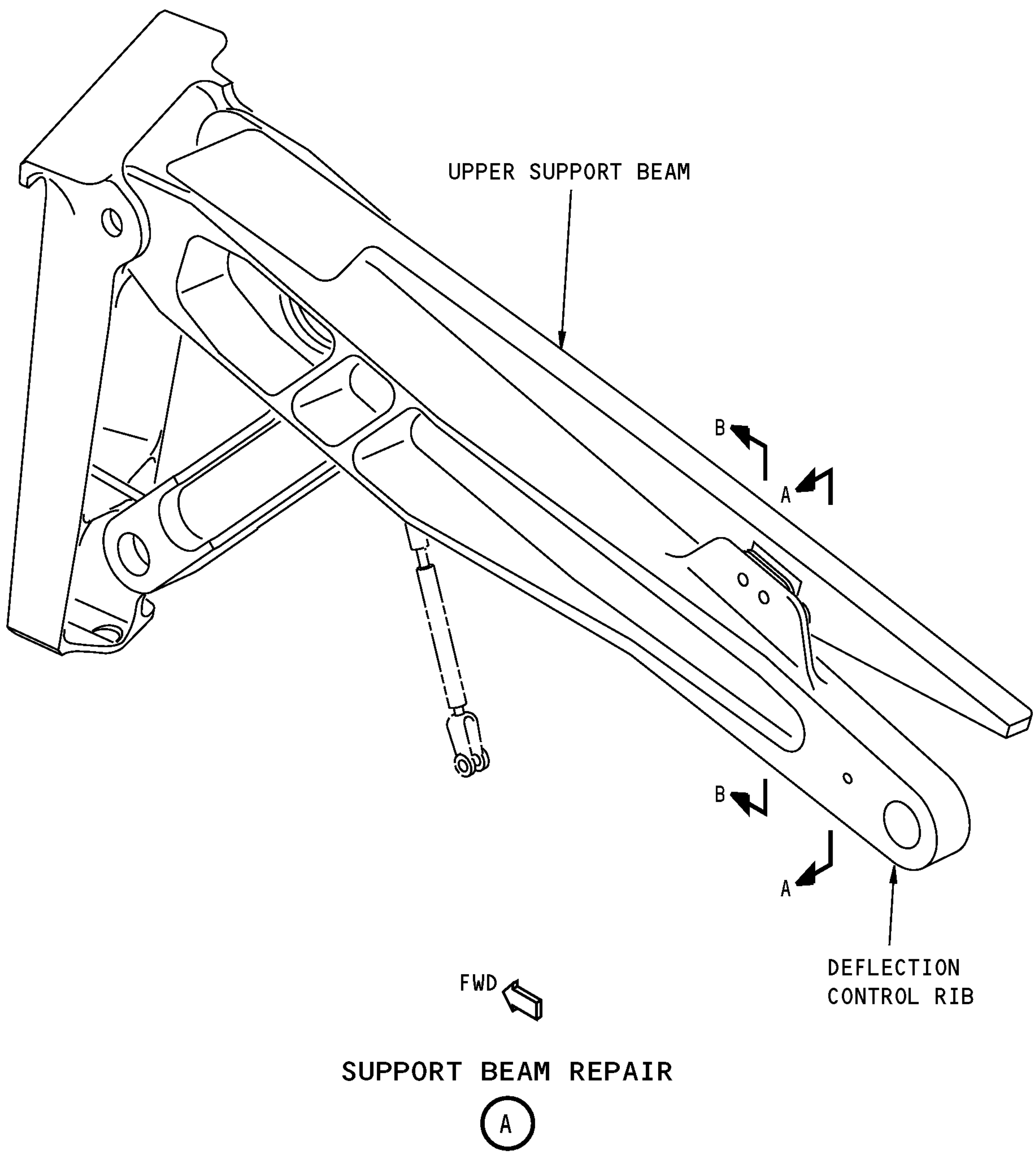
(1) CAD plate as given in SOPM 20-42-05.

(2) One layer of BMS 10-11, Type I primer and one layer of BMS 10-11, Type II enamel. Refer to SOPM 20-41-02.

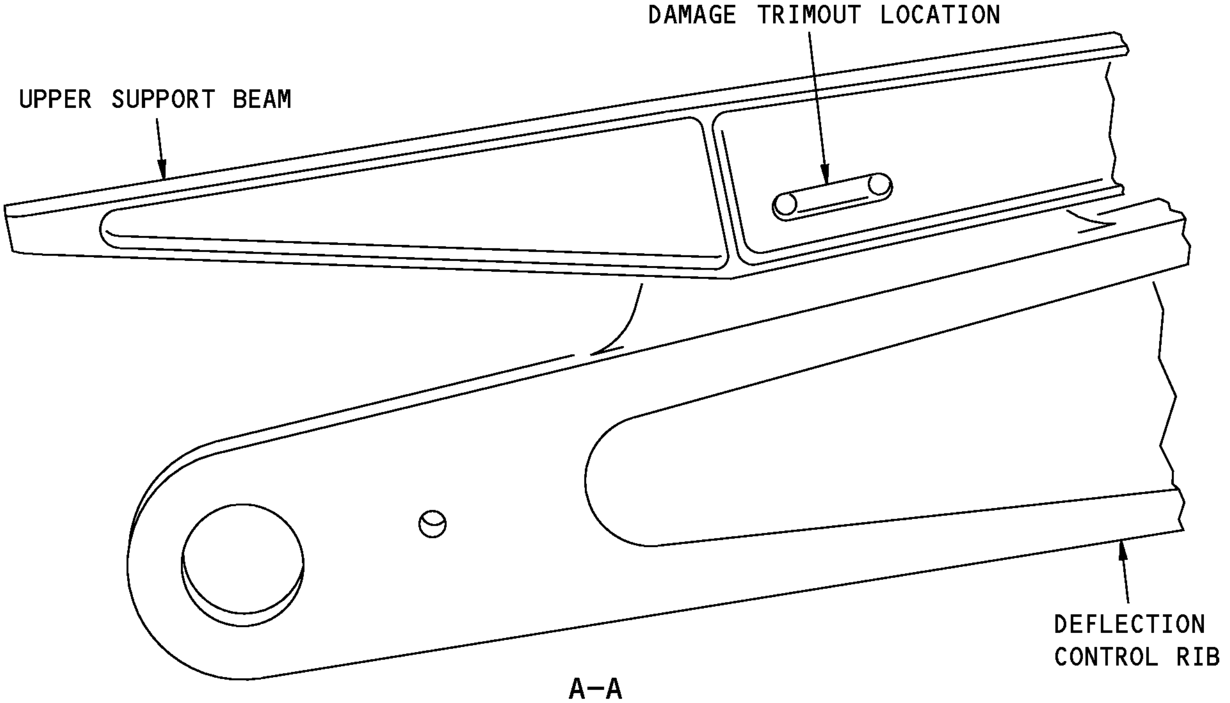
I. Finish the solid aluminum spacer as given in Figure 203, Sheet 1 with one layer of BMS 10-11, Type I primer and one layer of BMS 10-11, Type II enamel. Refer to SOPM 20-41-02.

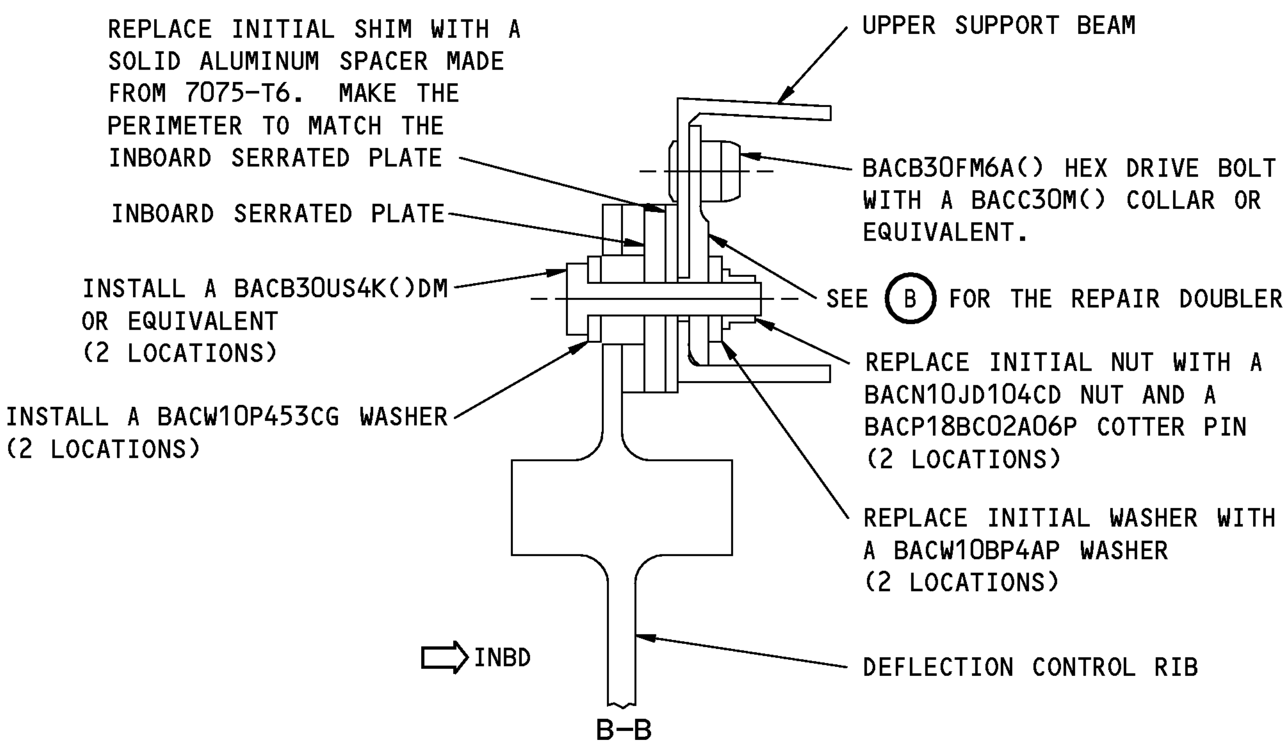
J. Fay surface seal between the part [1] repair doubler and the mating surfaces with BMS 5-95 sealant. Install the part [1] repair doubler and fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL. Make sure there is a 0.005 inch (0.13 mm) or less gap prior to fastener installation. Provide an edge seal around the repair slot in the support beam with BMS 5-95 sealant. 51-20-05, GENERAL.

(1) Make sure you install the 0.25 inch (6.35 mm) diameter BACB30US4K()DM or equivalent as shown in Figure 203.

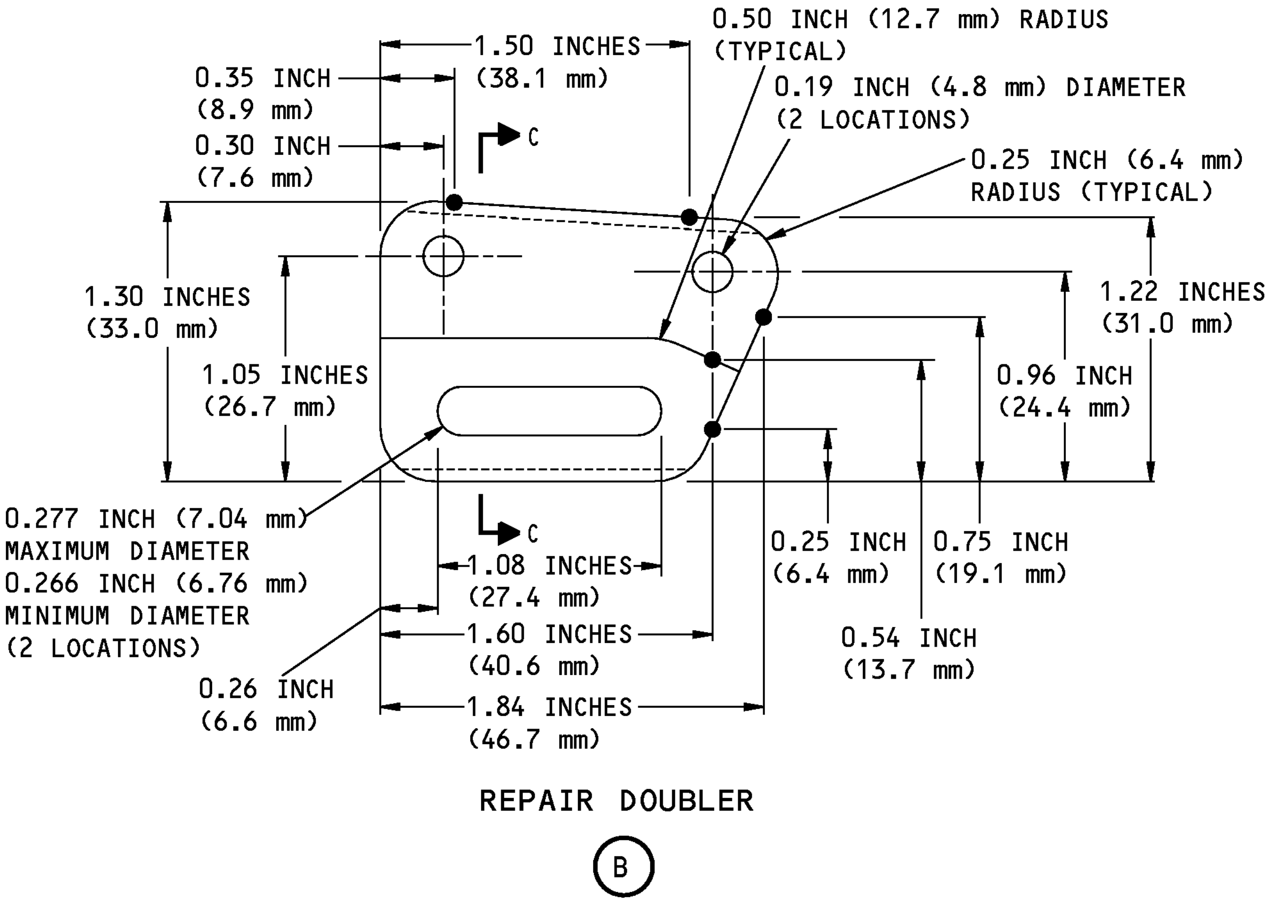


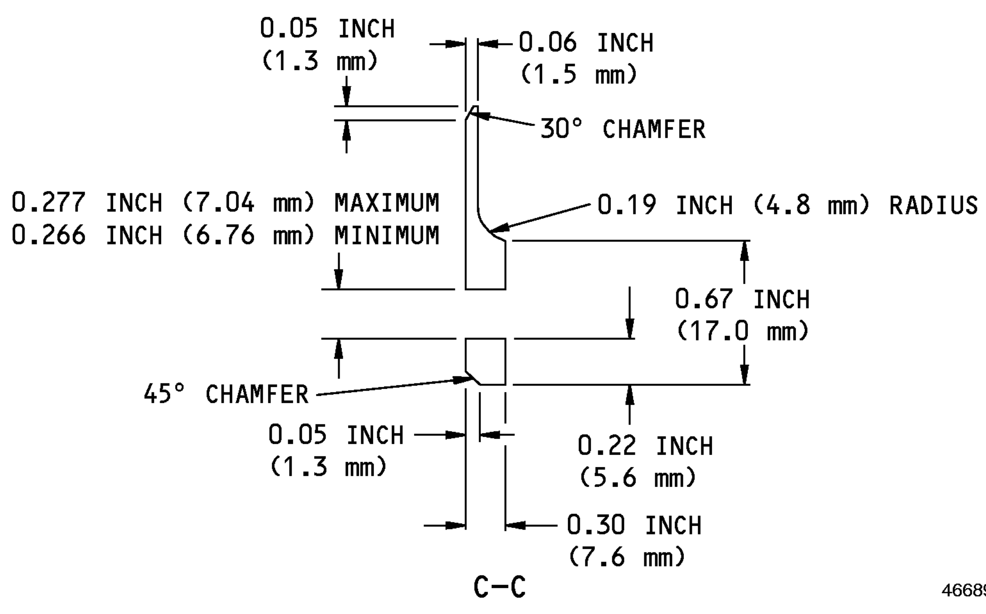

**Inboard Fixed Trailing Edge Support Beam Repair**
**Figure 203 (Sheet 1 of 3)**




**Inboard Fixed Trailing Edge Support Beam Repair**
**Figure 203 (Sheet 2 of 3)**

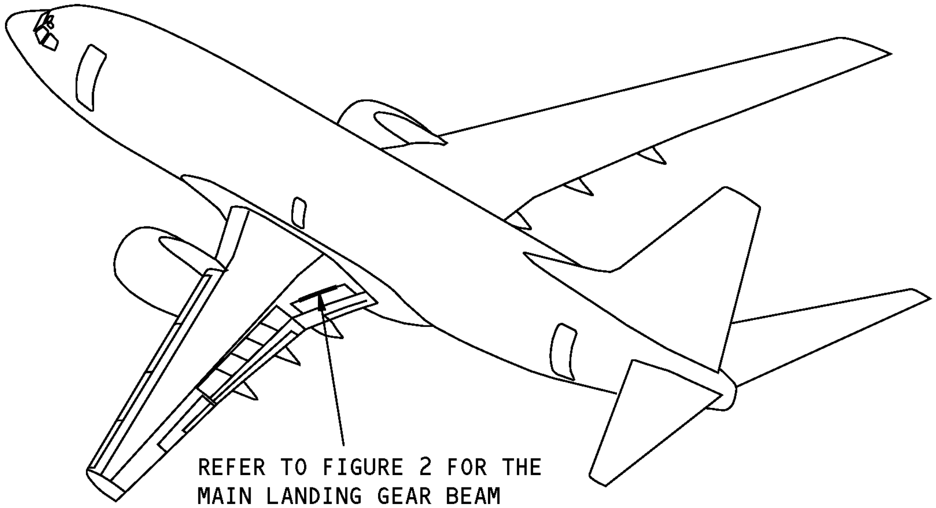



466896 S0000142389_V6

**Inboard Fixed Trailing Edge Support Beam Repair**
**Figure 203 (Sheet 3 of 3)**

## IDENTIFICATION 1 - MAIN LANDING GEAR BEAM AND STABILIZER STRUCTURE

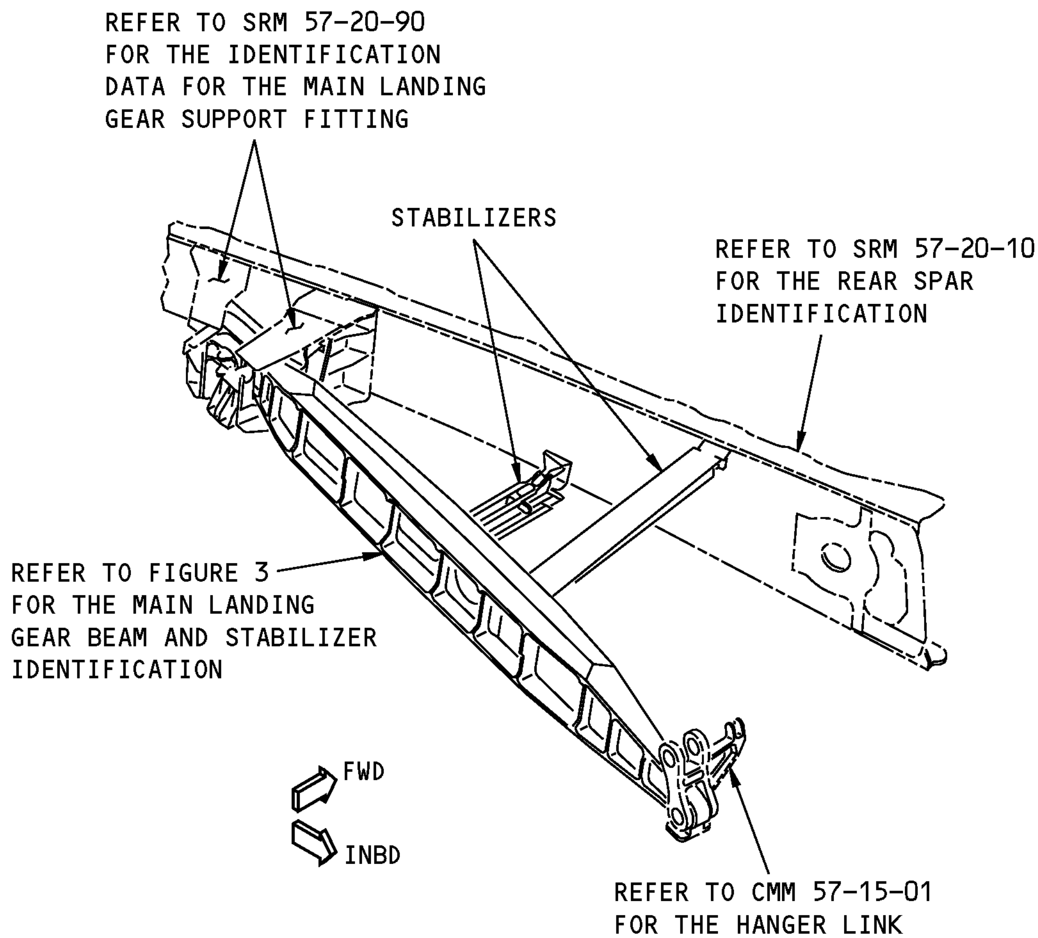



NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

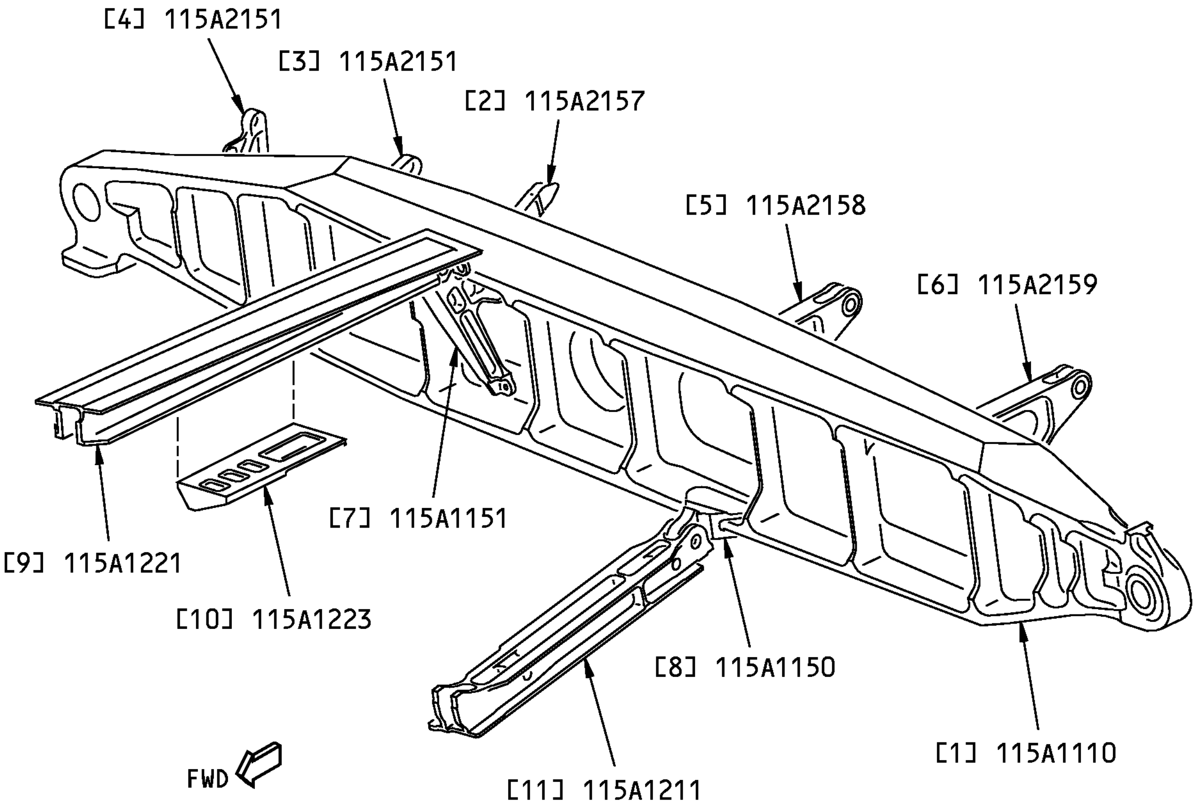
**Main Landing Gear Beam and Stabilizer**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 115A0001 | Functional Product Collector, Fixed Trailing Edge - Left Wing |
| 115A0002 | Functional Product Collector, Fixed Trailing Edge - Right Wing |
| 115A1100 | Main Landing Gear Beam Assembly |




**Main Landing Gear Beam and Stabilizer Location**
**Figure 2**




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Main Landing Gear Beam and Stabilizer Identification**
**Figure 3**

**Table 2**:

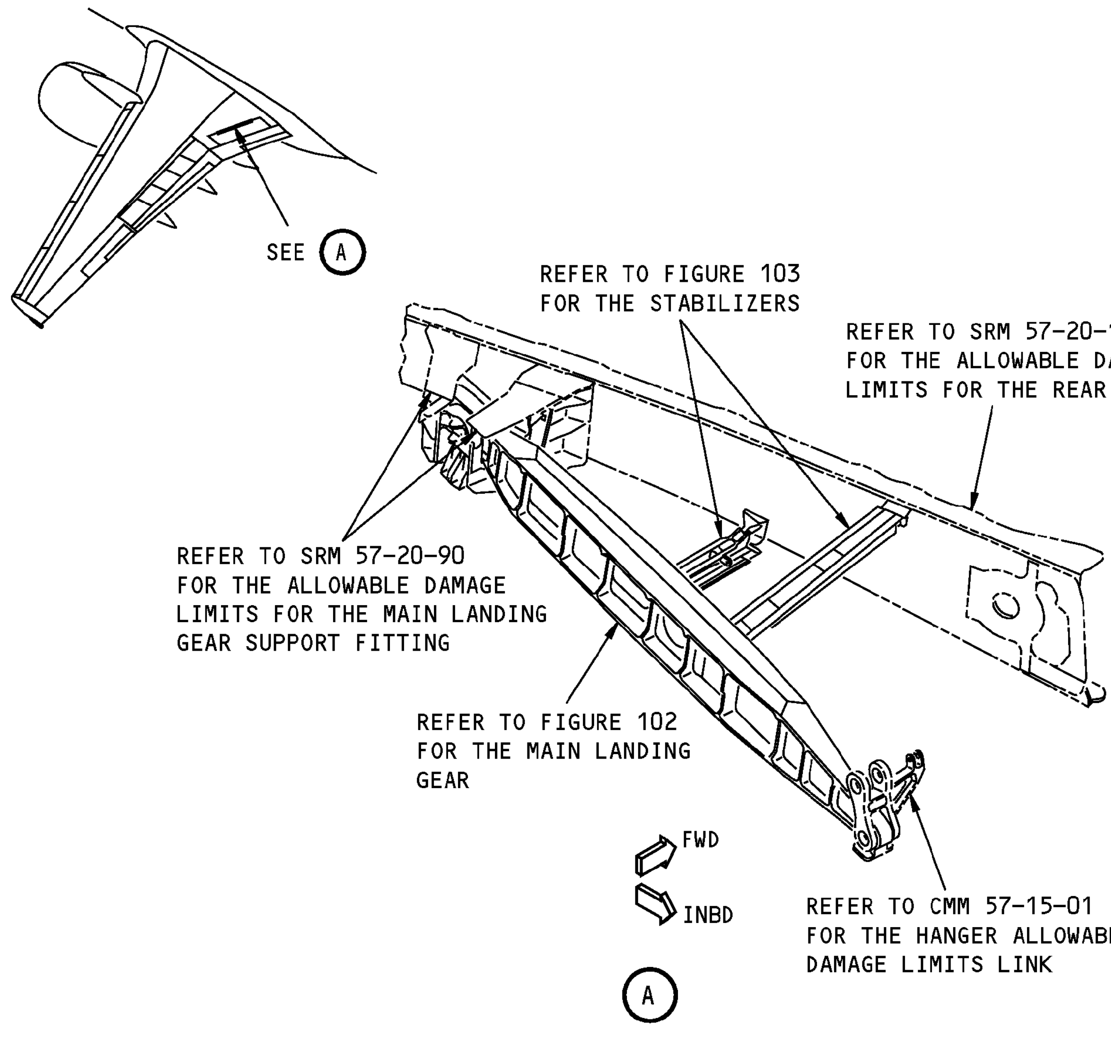
| LIST OF MATERIALS FOR FIGURE 3 | | | |
|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** |
| [1] | Main Landing Gear Beam | | Ti-6Al-4V forged titanium as given in BMS 7-247, Condition A |
| [2] | Hinge Fitting - Inboard Spoiler Support - MLG Beam | 2.000 (50.8) | 7050-T7451 plate as given in AMS 4050 (Grain direction controlled part) |
| [3] | Support Bracket | 1.500 (38.1) | 7050-T7451 plate as given in AMS 4050 (Grain direction controlled part) |
| [4] | Support Bracket, | 1.500 (38.1) | 7050-T7451 plate as given in AMS 4050 (Grain direction controlled part) |
| [5] | Spoiler Support Fitting - MLG Beam | 4.250 (107.95) | 7050-T7451 plate as given in BMS 7-323, Type III |
| [6] | Spoiler Support Fitting - MLG Beam | 5.000 (127) | 7050-T7451 plate as given in BMS 7-323, Type III |
| [7] | Upper Stabilizer Link Support | 5.500 (139.7) | Ti-6Al-4V forged titanium block as given in MIL-T-9047, annealed |
| [8] | Lower Stabilizer Support Fitting | 5.250 (133.35) | 7050-T7451 plate as given in BMS 7-323, Type I (Grain direction controlled part) |
| [9] | Upper Stabilizer Link Fitting | 2.750 (69.85) | 7050-T7451 plate as given in AMS 4050 |
| [10] | Upper Stabilizer System Support Fitting | 1.750 (44.45) | 7050-T7451 plate as given n AMS 4050 (Grain direction controlled part) |
| [11] | Lower Stabilizer Link | 4.300 (109.22) | 7050-T7451 plate as given in BMS 7-323, Type III (Grain direction controlled part) |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## ALLOWABLE DAMAGE 1 - MAIN LANDING GEAR BEAM INSTALLATION

### 1. Applicability

A. This subject gives the allowable damage limits for the main landing gear beam and the beam stabilizers as shown in Main Landing Gear Beam Installation, Figure 101/ALLOWABLE DAMAGE 1.




**Main Landing Gear Beam Installation**
**Figure 101**

### 2. General

A. Remove the parts as necessary to get access to the damaged area.

B. Refer to Main Landing Gear Beam Zones, Figure 102/ALLOWABLE DAMAGE 1 for the allowable damage zones of the beam. Refer to Main Landing Gear Beam Stabilizers, Figure 103/ALLOWABLE DAMAGE 1 for the beam stabilizers.

**WARNING:** SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE ON THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STEAM EXPLOSION.

C. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of abrasives and other materials you can use to remove damage.

(3) Refer to 51-30-05 for possible sources of equipment and tools you can use to remove damage.

(4) Make sure that all the damage has been removed within the limits. Refer to SOPM 20-20-02 for penetrant inspection procedures or 737 NDT Part 6, 51-00-00, Figure 4 for eddy current inspection procedures.

D. After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the beam and beam stabilizers.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

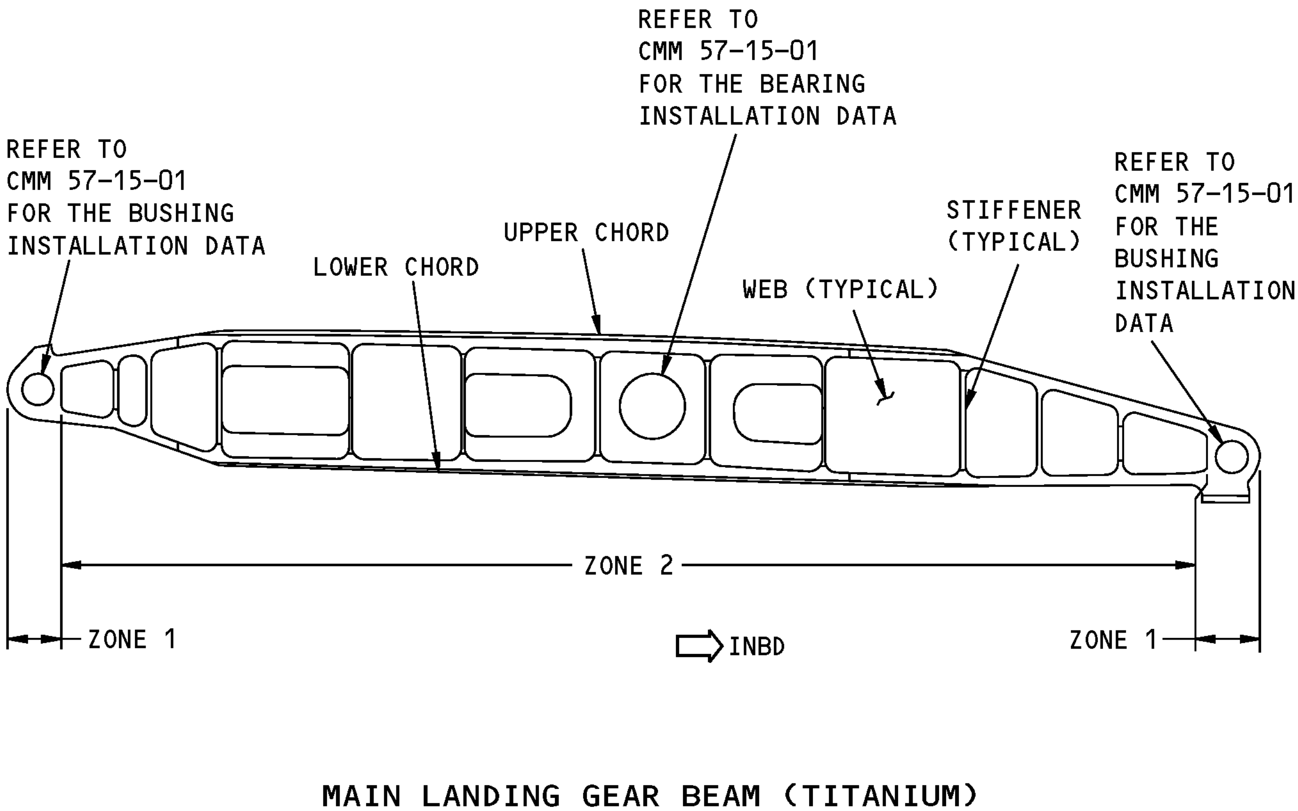
1) Use an intensity 0.010A to 0.021A for the beam.

2) Use an intensity of 0.005A to 0.010A for the stabilizers.

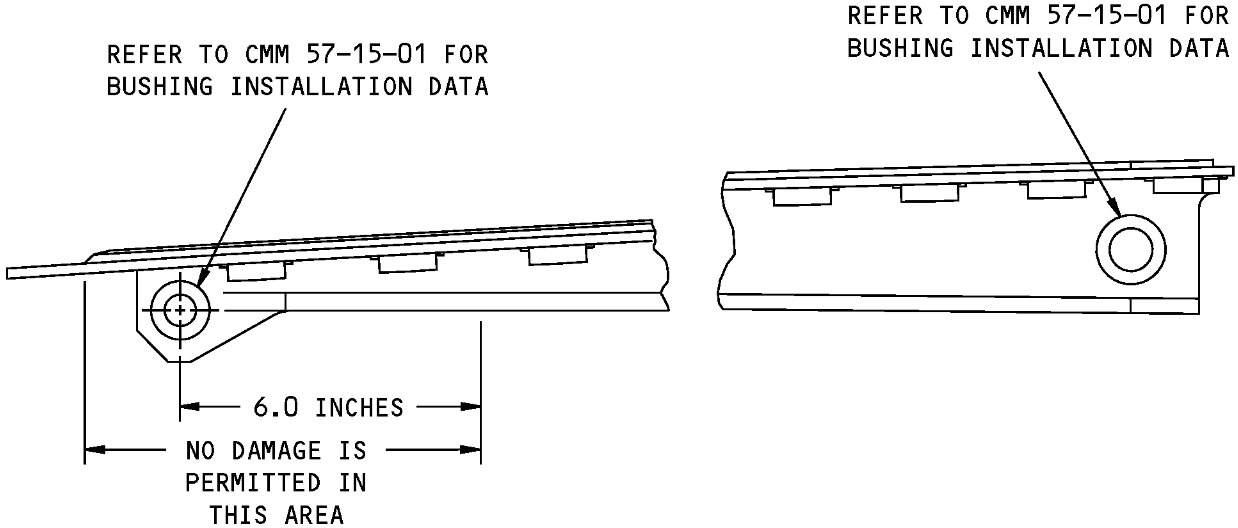
3) Use coverage 2.0.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked aluminum parts (stabilizers only). Refer to 51-20-01.

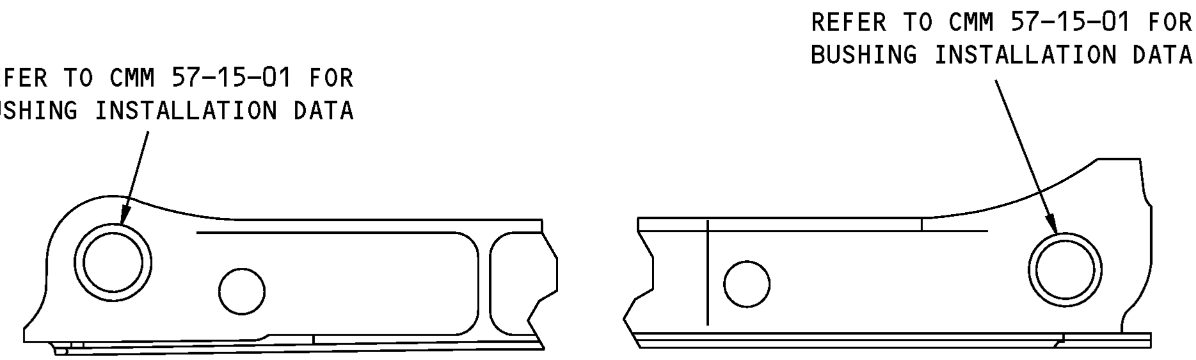
(3) Apply a layer of BMS 10-11, Type I primer to the surface of the reworked area. Refer to SOPM 20-41-02.




**Main Landing Gear Beam Zones**
**Figure 102**




UPPER STABILIZER LINK
(ALUMINUM)




LOWER STABILIZER LINK
(ALUMINUM)

**Main Landing Gear Beam Stabilizers**
**Figure 103**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| CMM 57-15-01, | Main Landing Gear Beam Installlation Components |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 4 | Surface Inspection of Aluminum Parts |

## 4. Allowable Damage Limits

A. Main Landing Gear Beam

(1) Zone 1

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail A.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Zone 2 - Upper Chord

(a) Damage is not permitted.

(3) Zone 2 - Lower Chord, Web and Stiffeners

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details B and C.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

B. Stabilizers

(1) Cracks:

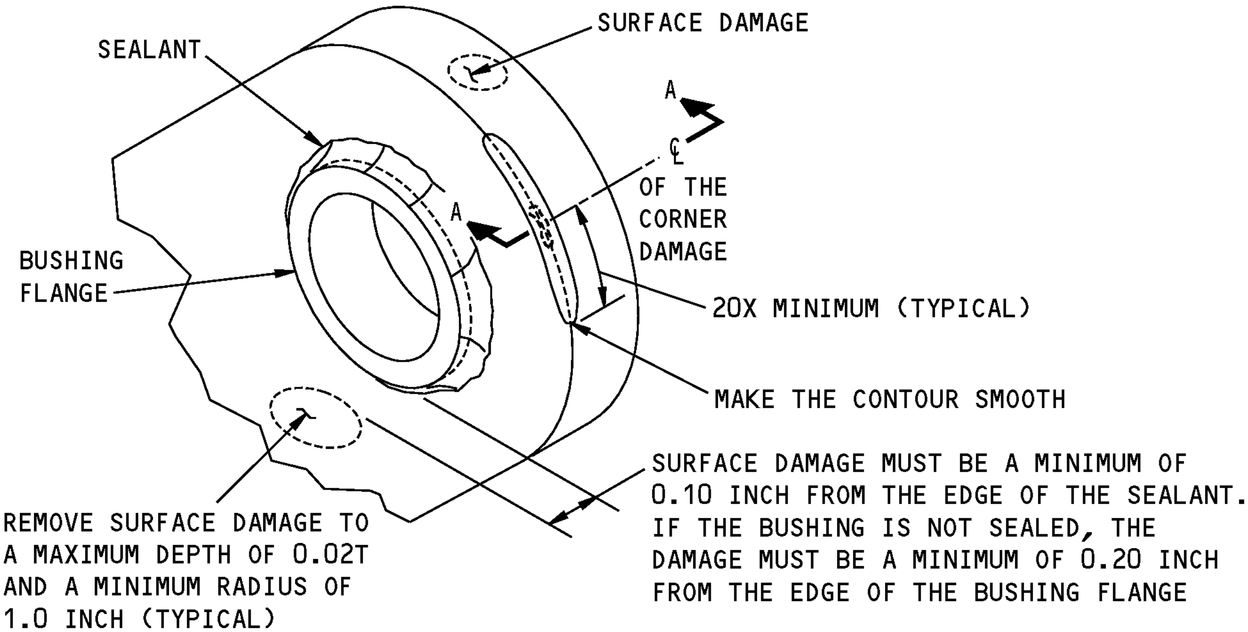
(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B and D.

(3) Dents are not permitted.

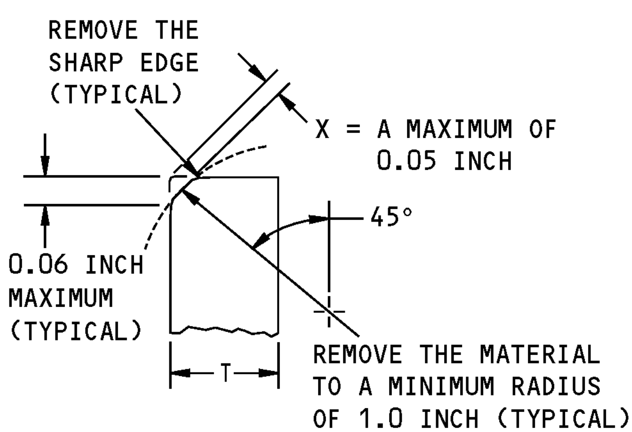
(4) Holes and Punctures are not permitted.




NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

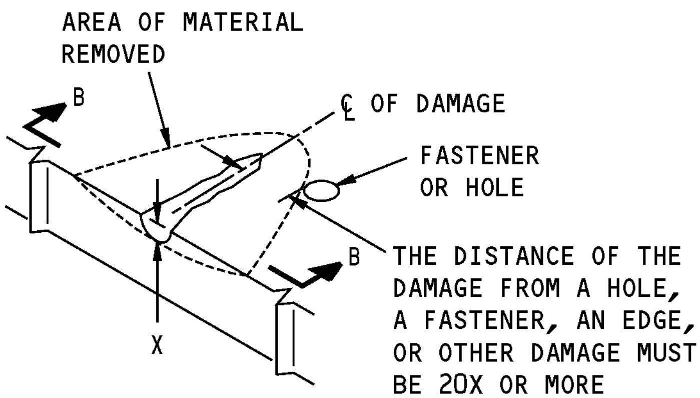
REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

A

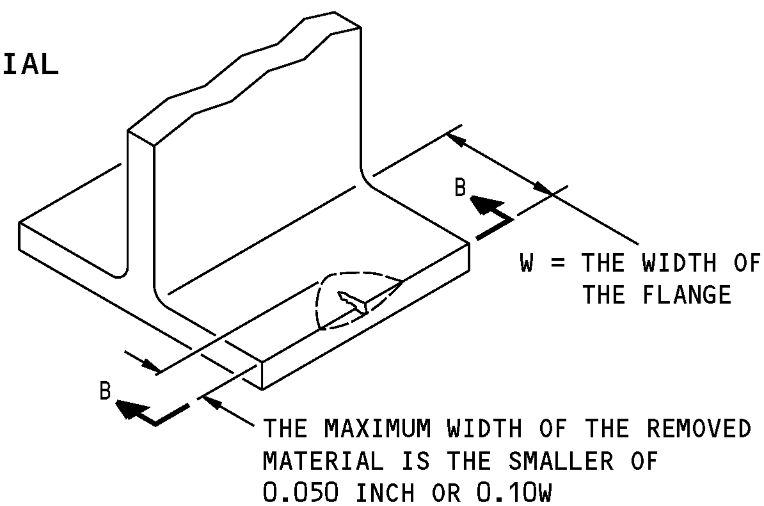



A-A

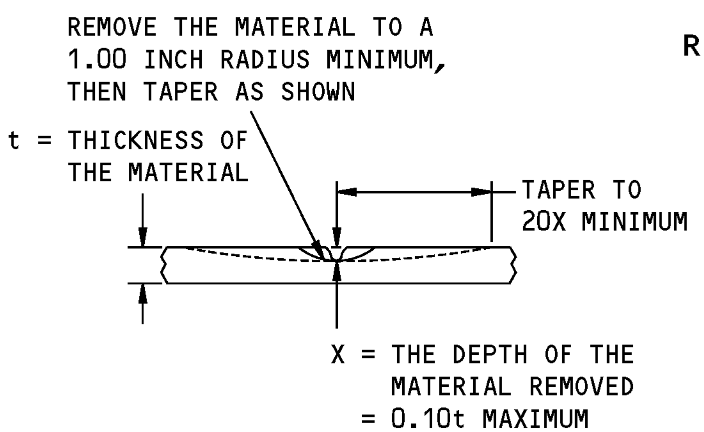
**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**




REMOVAL OF DAMAGED MATERIAL ON A SURFACE
B

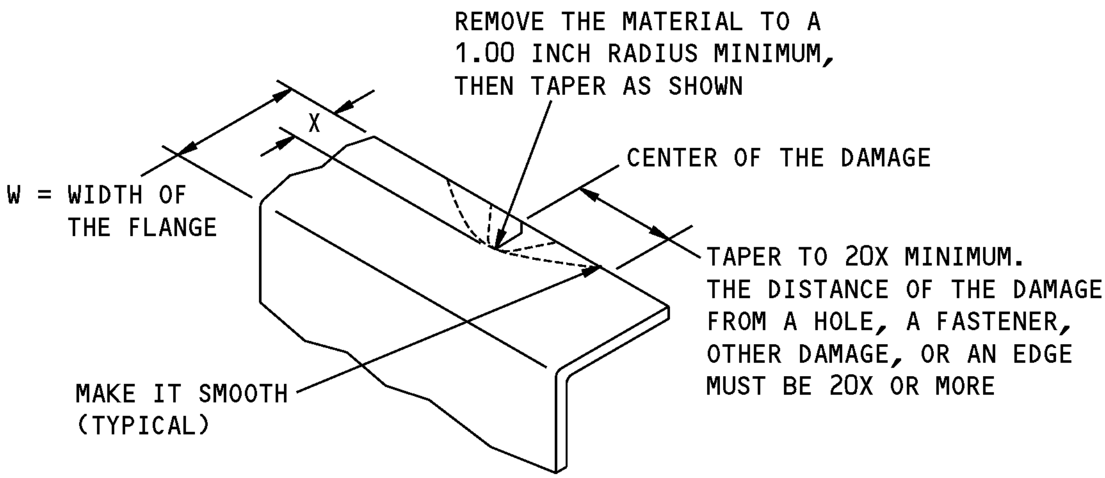



REMOVAL OF DAMAGED MATERIAL ON A SURFACE
C




B–B

**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**



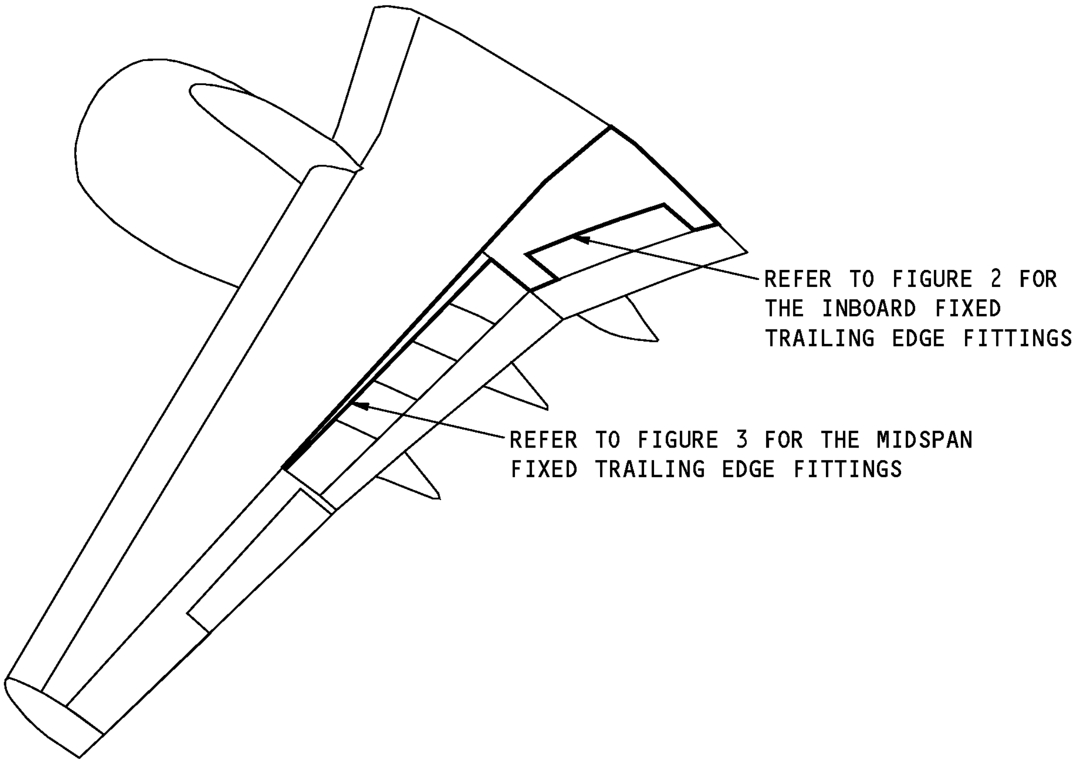

X = THE WIDTH OF THE MATERIAL REMOVED
= 0.10W MAXIMUM

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**

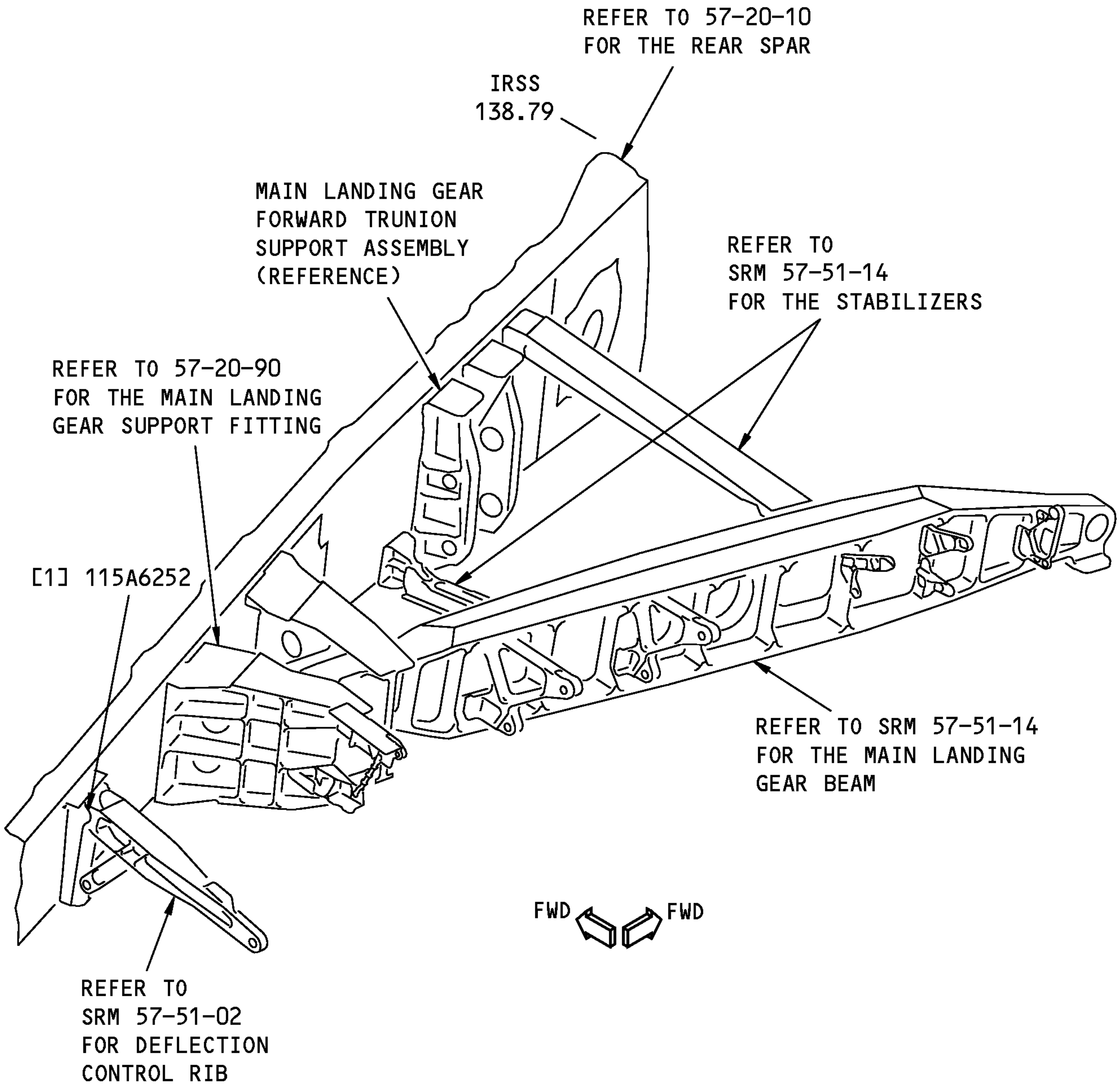
## IDENTIFICATION 1 - WING TRAILING EDGE FITTINGS




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Trailing Edge Location**
**Figure 1**

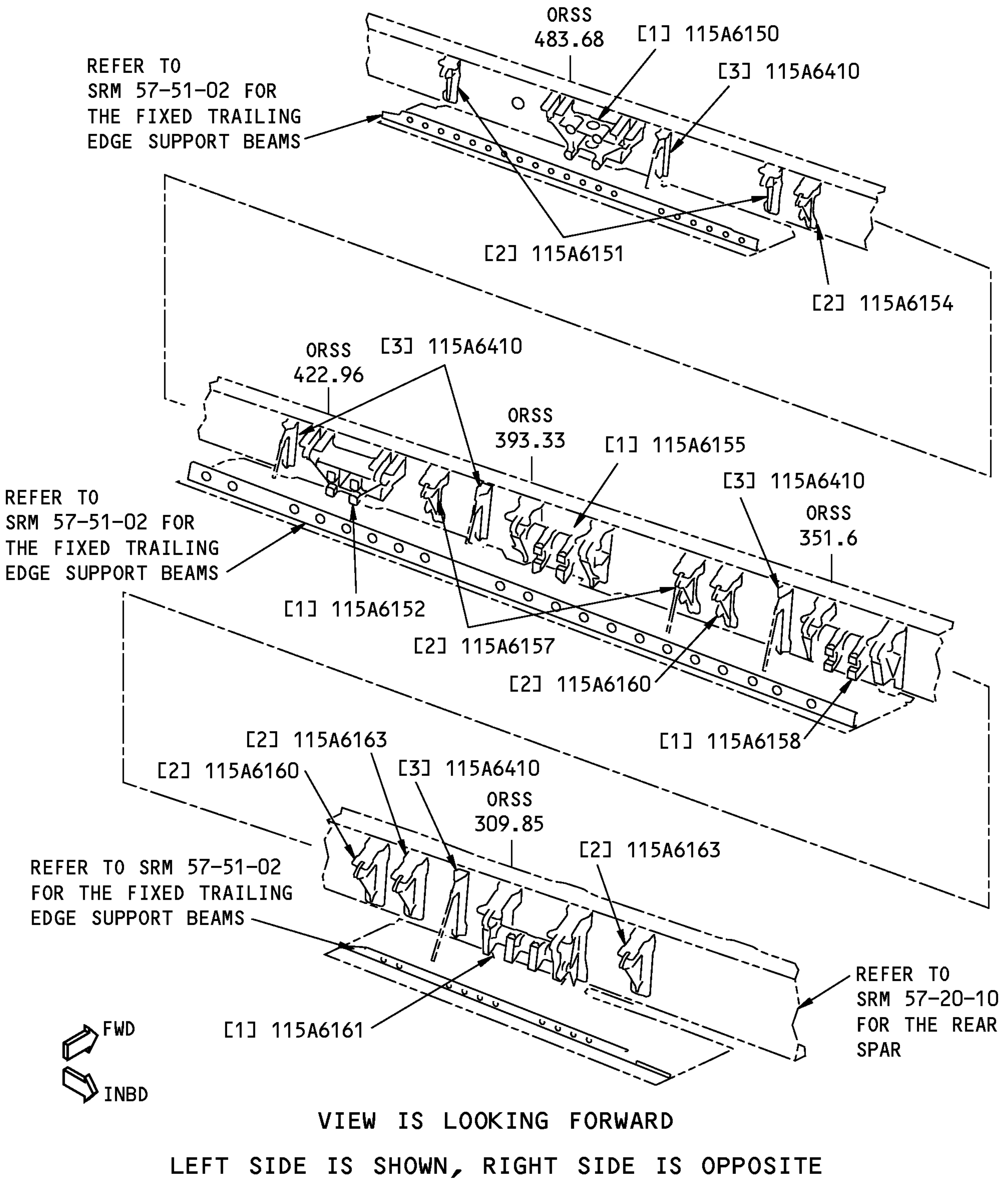
| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 115A0001 | Functional Product Collector, Fixed Trailing Edge - Left Wing |
| 115A0002 | Functional Product Collector, Fixed Trailing Edge - Right Wing |
| 115A6100 | Hinge/Actuator Support Installation - Outboard Spoilers, Midspan |
| 115A6200 | Rib Installation - Deflection Control, Midspan Fixed Trailing Edge |
| 115A6400 | Rod Support Installation - Midspan Fixed Trailing Edge |




**Inboard Wing Trailing Edge Fittings**
**Figure 2**

| LIST OF MATERIALS FOR FIGURE 2 | | | |
|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL |
| [1] | Deflection Control Rib Attach Fitting | | 7050-T7451 machined plate as given in BMS 7-323. |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Midspan Wing Trailing Edge Fittings**
**Figure 3**

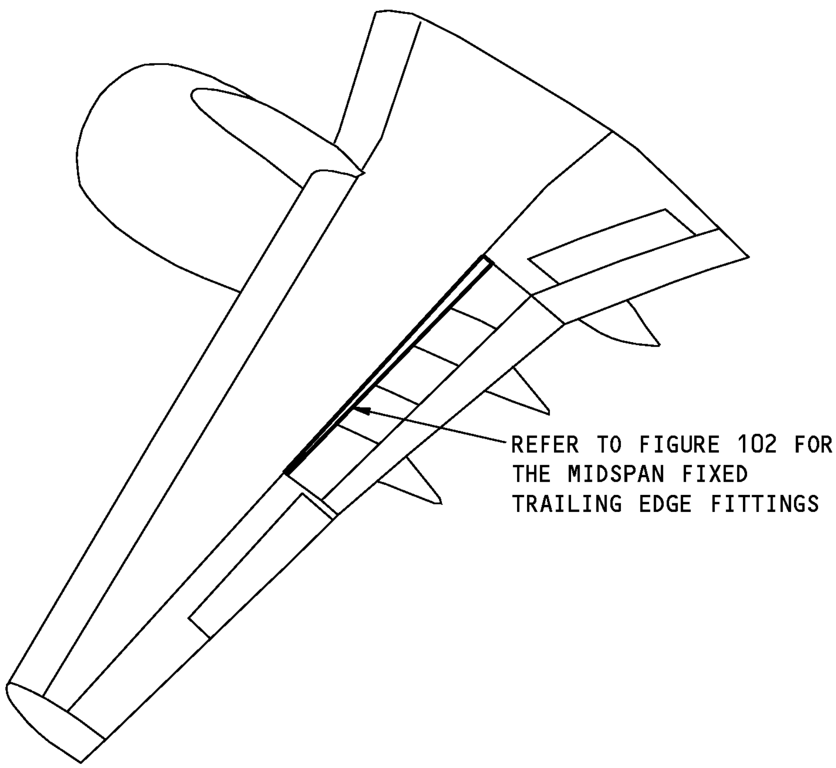
| LIST OF MATERIALS FOR FIGURE 3 | | | |
|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL |
| [1] | Spoiler Hinge/Actuator Support Fitting | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [2] | Spoiler Hinge Support Fitting | | 7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part. |
| [3] | Rod Support Assembly | | 7050-T7451 machined plate as given in AMS 4050. |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

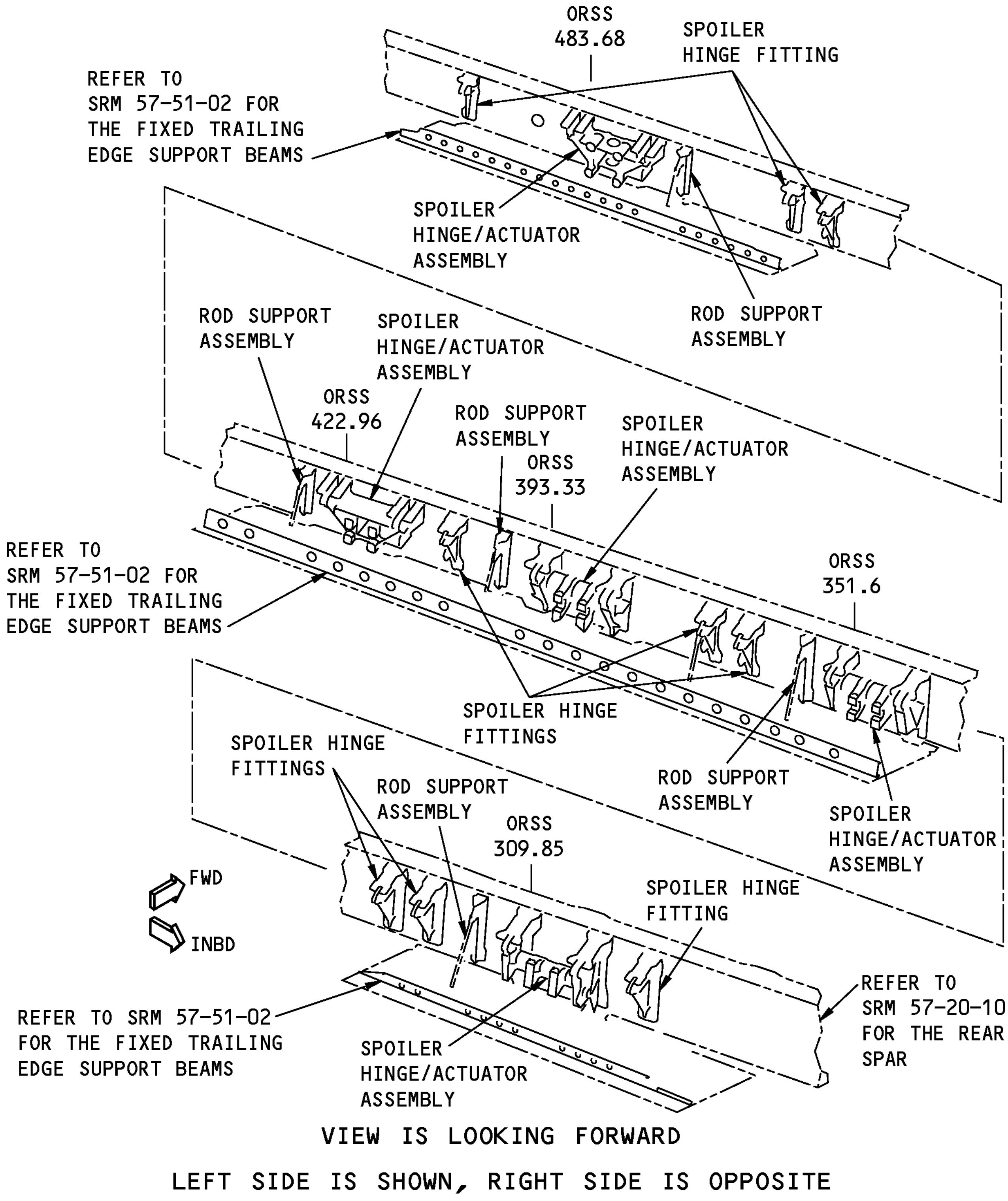
## ALLOWABLE DAMAGE 1 - WING TRAILING EDGE FITTINGS

### 1. Applicability

A. Allowable Damage 1 is applicable to damage on the wing fixed trailing edge fittings shown in Figure 101/ALLOWABLE DAMAGE 1.




**Wing Trailing Edge Location**
**Figure 101**




**Midspan Wing Trailing Edge Fittings**
**Figure 102**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limit references that are applicable to each type of structure.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| TYPE OF STRUCTURE | PARAGRAPH |
| ROD SUPPORT ASSEMBLY | 4.A |
| SPOILER HINGE/ACTUATOR FITTING ASSEMBLY | 4.B |
| SPOILER HINGE FITTING | 4.B |

C. Remove the damage.

(1) Refer to 51-10-02, GENERAL for the investigation and cleanup procedures.

(2) Refer to 51-30-03, GENERAL for possible sources of the abrasive and other materials you can use to remove the damage.

(3) Refer to 51-30-05, GENERAL for possible sources of the equipment and tools you can use to remove the damage.

D. After you remove the damage, do as follows:

**WARNING:** WEAR EYE PROTECTION WHEN YOU USE THE ROTARY PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the parts given in Table 102/ALLOWABLE DAMAGE 1.

(a) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

**Table 102**:

| PARTS THAT YOU MUST SHOT PEEN OR FLAP PEEN AFTER YOU REMOVE THE DAMAGE | |
|---|---|
| PART DRAWING NUMBER | PART NAME |
| 115A6130 | Hinge/Actuator Support Assembly - Spoiler No. 1 |
| 115A6132 | Hinge/Actuator Support Assembly - Spoiler No. 2 |
| 115A6133 | Hinge Fitting Assembly - Spoiler No. 2 |
| 115A6134 | Hinge/Actuator Support Assembly - Spoiler No. 3 |
| 115A6136 | Hinge/Actuator Support Assembly - Spoiler No. 4 |
| 115A6138 | Hinge/Actuator Support Assembly - Spoiler No. 5 |
| 115A6410 | Rod Support Assembly |

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01, GENERAL.

(3) Apply two layers of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-44-04.

## 3. References

| Reference | Title |
| --- | --- |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01, GENERAL | Protective Treatment of Metallic and Composite Materials |
| 51-20-06, GENERAL | Shot Peening |
| 51-30-03, GENERAL | Sources for Non-Metallic Repair Materials |
| 51-30-05, GENERAL | Equipment and Tools For Repairs |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Rod Support Assemblies

**NOTE**: Damage on the edges of large holes and cutouts is not permitted.

(1) Cracks:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A and B.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Spoiler Hinge/Actuator Support Assemblies and Spoiler Hinge Fitting Assemblies

**NOTE**: Damage on the edges of large holes and cutouts is not permitted.

(1) Cracks:

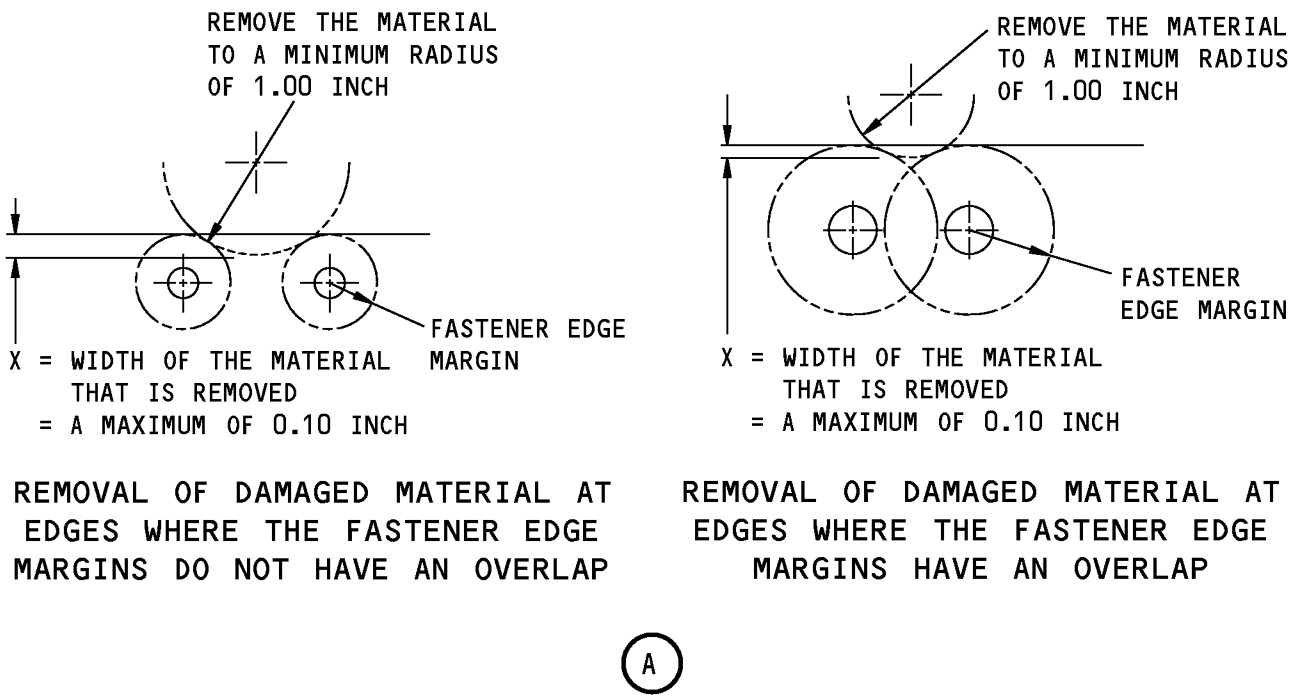
(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A and B.

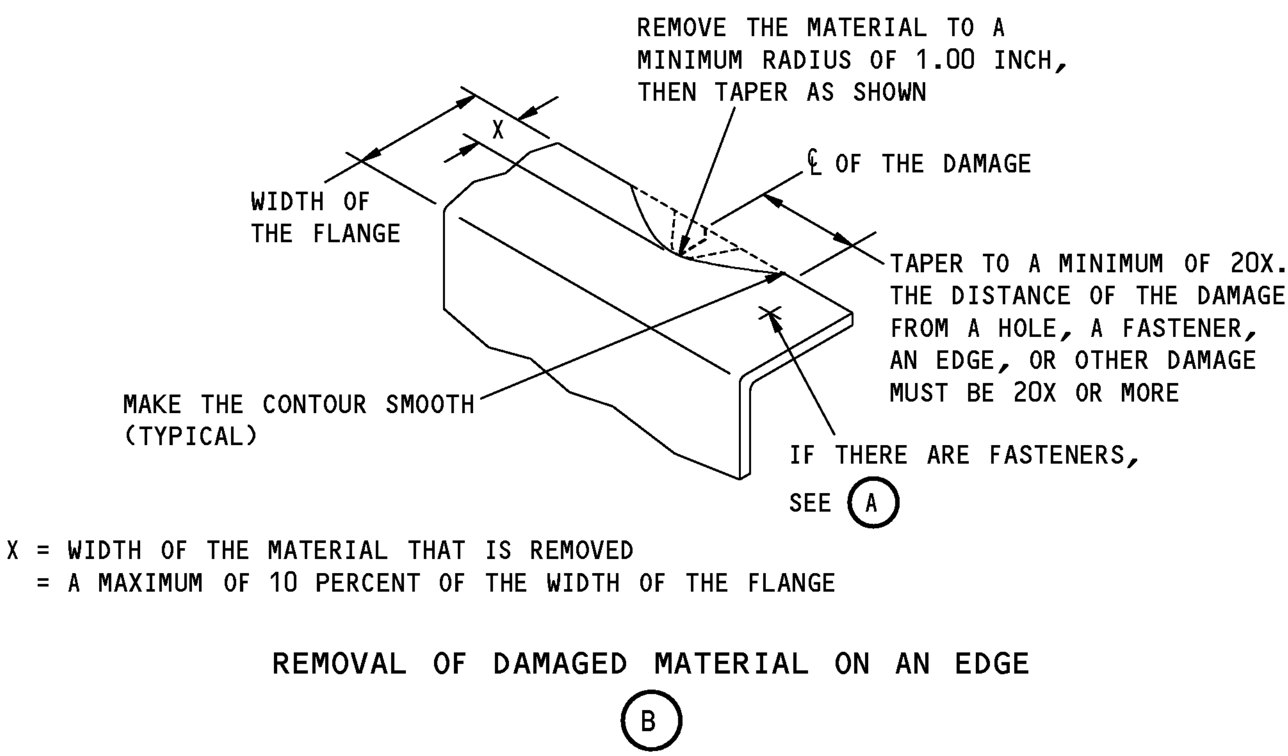
(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

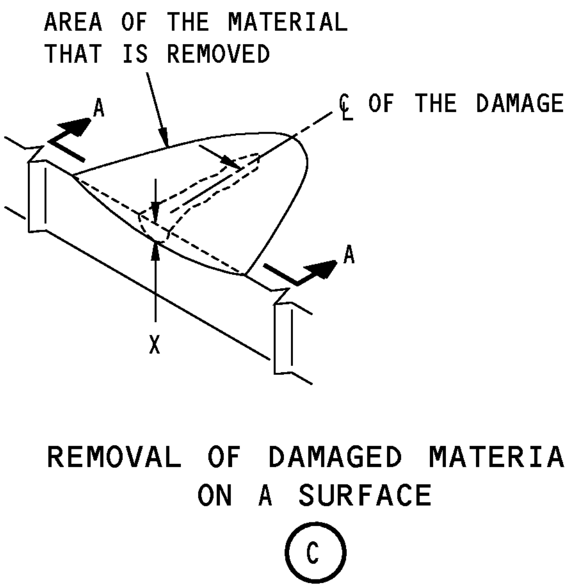
(3) Dents are not permitted.

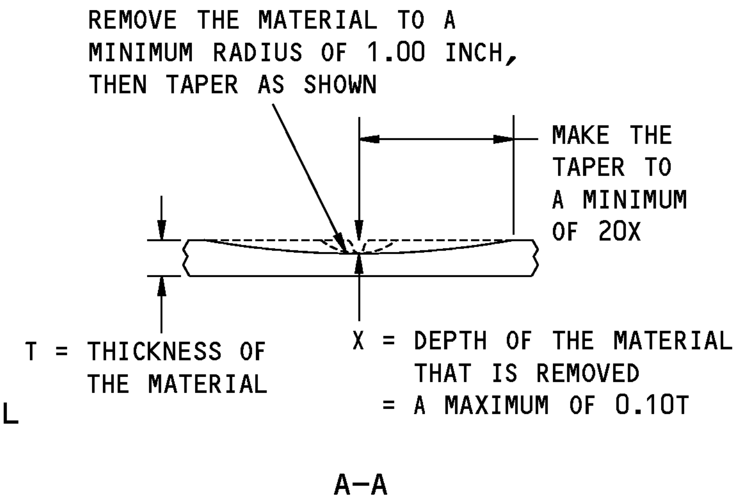
(4) Holes and Punctures are not permitted.

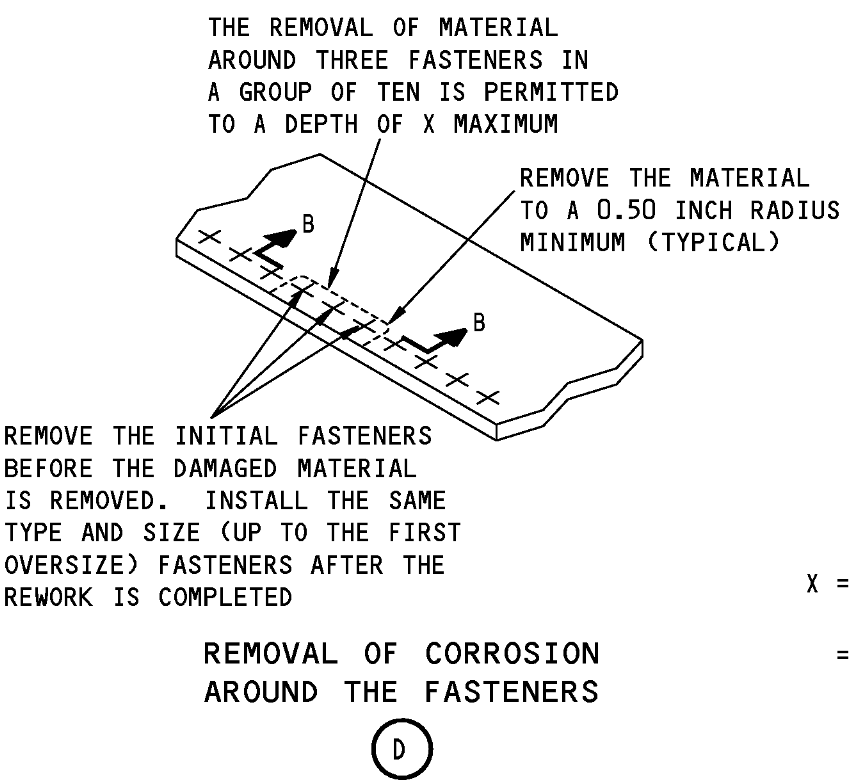


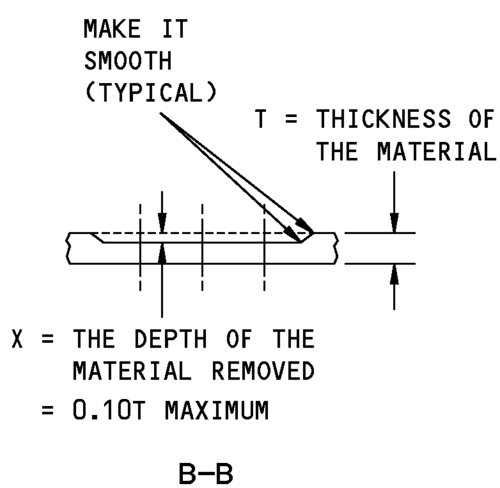




**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 2)**



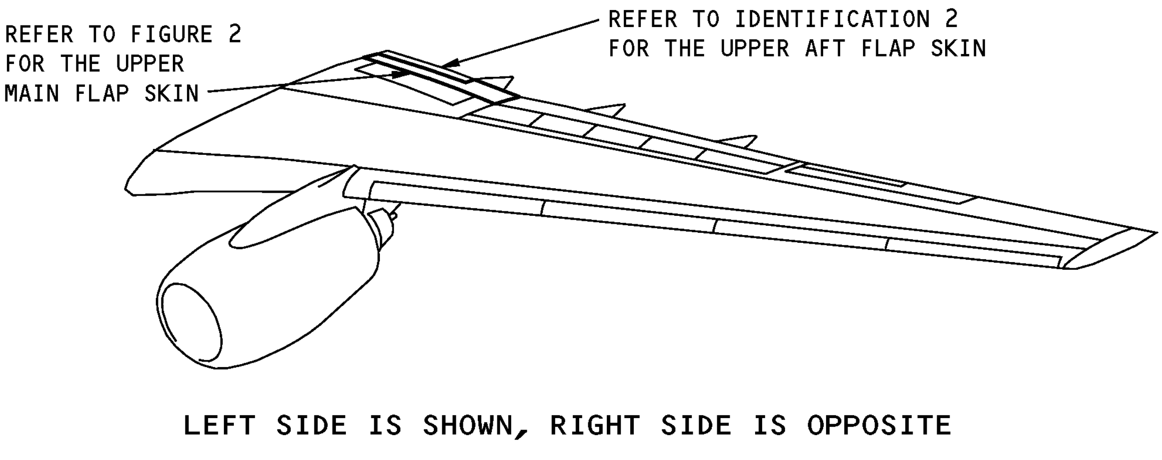



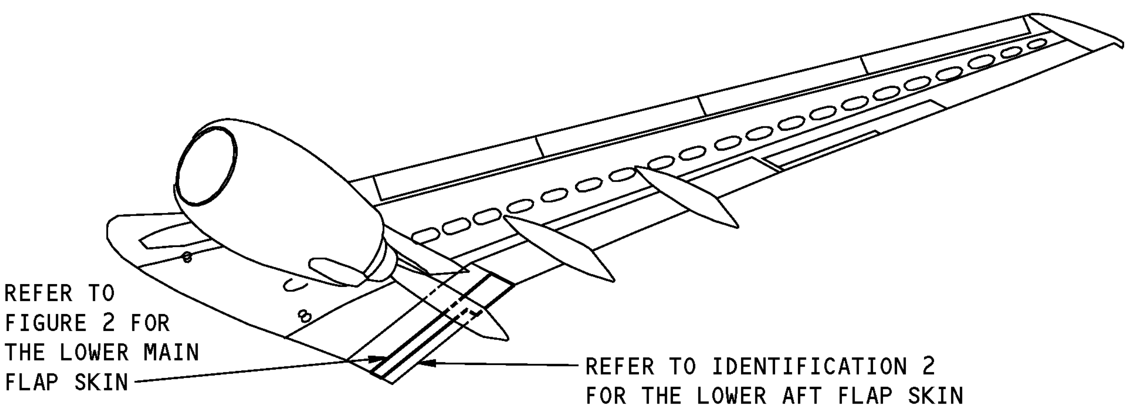




**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 2)**

## IDENTIFICATION 1 - WING TRAILING EDGE INBOARD MAIN FLAP SKIN



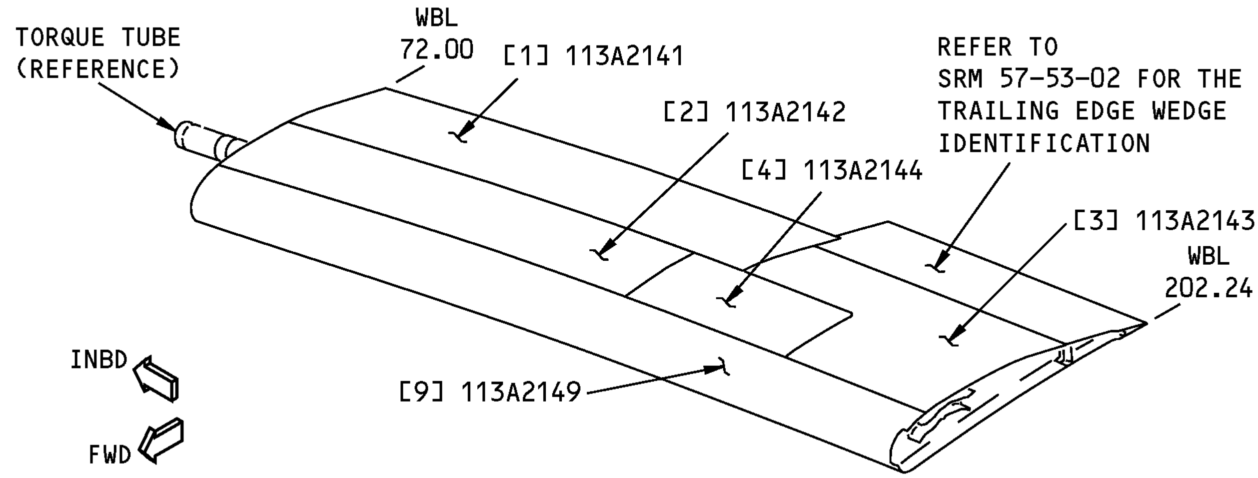




**Wing Trailing Edge - Inboard Main Flap Skin Panel Location**
**Figure 1**

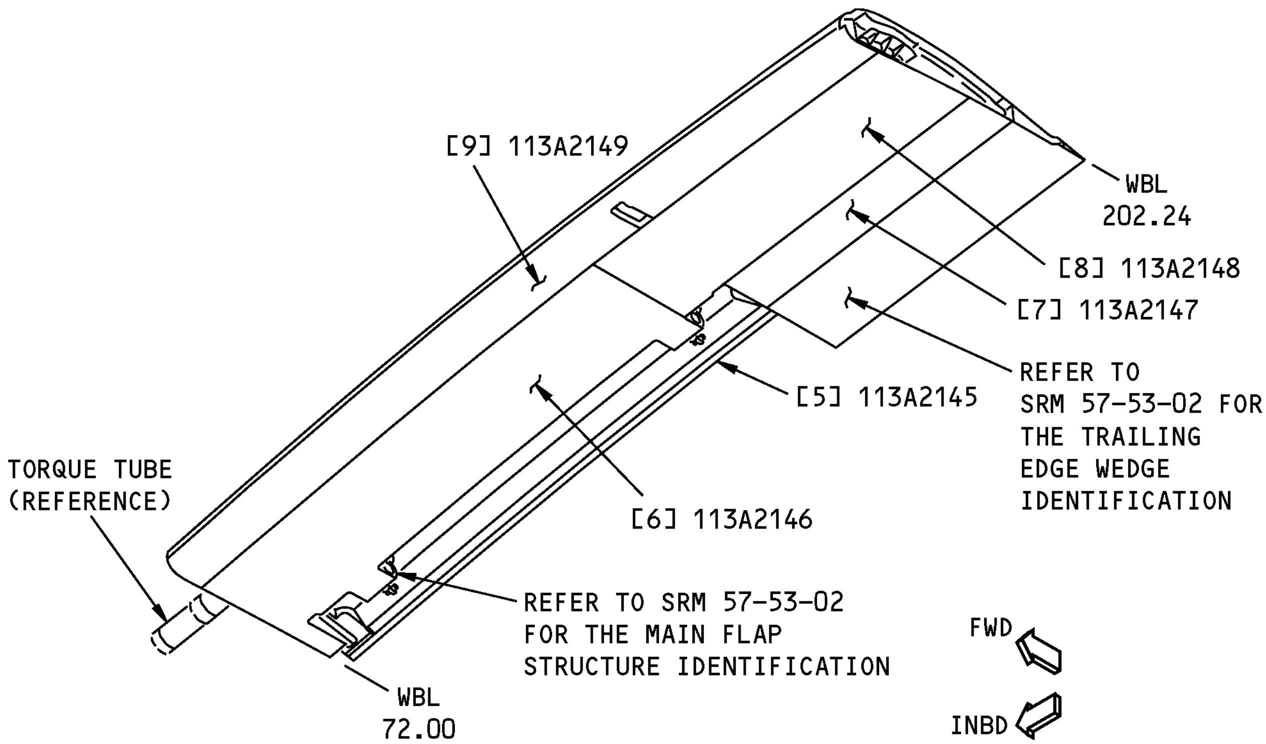
**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A2001 | Flap Installation - Inboard Trailing Edge Flap |
| 113A2002 | Flap Assembly - Inboard Trailing Edge Flap |
| 113A2100 | Main Flap Assembly - Inboard Trailing Edge Flap |
| 113A2500 | Wedge Assembly - Bonded Part, Inboard Trailing Edge Main Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A2141 | Skin Panel Assembly - Bonded Part, Upper Trailing Edge Cove |
| 113A2142 | Skin Panel - Upper Inboard, Inboard Trailing Edge Main Flap |
| 113A2143 | Skin Panel - Upper Outboard, Inboard Trailing Edge Main Flap |
| 113A2144 | Skin Panel - Upper Middle, Inboard Trailing Edge Main Flap |
| 113A2145 | Skin Panel - Lower Trailing Edge Cove, Inboard Trailing Edge Main Flap |
| 113A2146 | Skin Panel - Lower Inboard, Inboard Trailing Edge Main Flap |
| 113A2147 | Skin Panel - Lower Outboard, Aft, Inboard Trailing Edge Main Flap |
| 113A2148 | Skin Panel - Lower Outboard, Inspar, Inboard Trailing Edge Main Flap |
| 113A2149 | Skin Panel - Leading Edge, Inboard Trailing Edge Main Flap |




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL



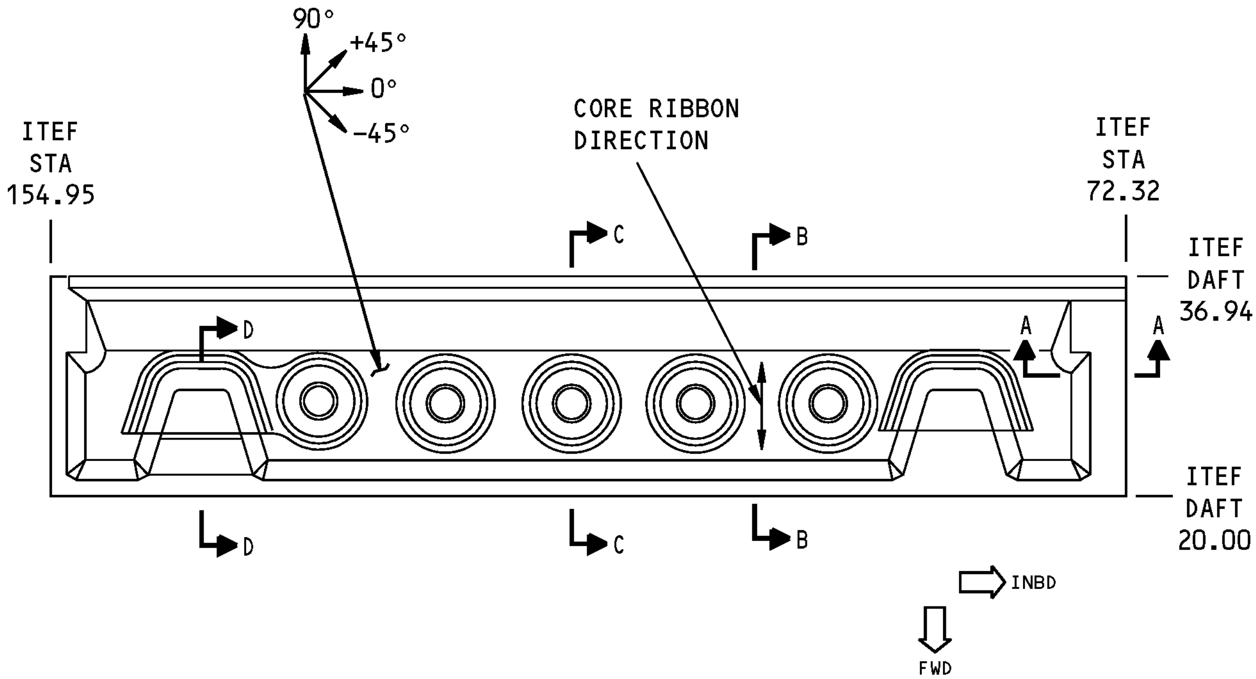

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

**Wing Trailing Edge - Inboard Main Flap Skin Panel Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Inboard Main Flap, Upper Trailing Edge Cove - Bonded Part | | Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW | |
| | Skin | | Refer to Figure 2 | |
| | Core | 0.500 | Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction. | |
| | Phenolic Strip | 0.025 | MIL-I-247628/14. | |
| [2] | Inboard Main Flap, Upper Inboard Skin | 0.080 | 2024-T3 bare sheet as given in QQ-A-250/4. | |
| [3] | Inboard Main Flap, Upper Outboard Skin - Bonded Part | | Bond with structural adhesive as given in BMS 5-101, Type II, Class I, Grade 10, as given in BAC 5514-5101. | |
| | Outer Skin | 0.100 | 7075-T62 bare sheet as given in QQ-A-250/12. | |
| | Inner Skin | 0.016 | 7075-T6 bare sheet as given in QQ-A-250/12. | |
| | Core | 0.500 | Aluminum honeycomb as given in BMS 4-4, Type 3-10, Class NPA. | |
| [4] | Inboard Main Flap, Upper Middle Skin | 0.125 | 2024-T3 bare sheet as given in QQ-A-250/4. | |
| [5] | Inboard Main Flap, Lower Trailing Edge Cove Skin | 0.125 | 7075-T6 bare sheet as given in QQ-A-250/12. | |
| [6] | Inboard Main Flap, Lower Inboard Skin | 0.125 | 2024-T3 bare sheet as given in QQ-A-250/4. | |
| [7] | Inboard Main Flap, Lower Outboard Aft Skin - Bonded Part | | Bond with structural adhesive as given in BMS 5-137, Type II, Class I, Grade 10, as given in BAC 5514-5137. | |
| | Outer Skin | 0.125 | 7075-T6 bare sheet as given in QQ-A-250/12. | |
| | Inner Skin | 0.016 | 7075-T6 bare sheet as given in QQ-A-250/12. | |
| | Core | 0.500 | Aluminum honeycomb as given in BMS 4-4, Type 4-25, Class NPA. | |
| [8] | Inboard Main Flap, Inspar Lower Outboard Skin - Bonded Part | | Bond with structural adhesive as given in BMS 5-101, Type II, Class I, Grade 10, as given in BAC 5514-5101. | |
| | Outer Skin | 0.080 | 2024-T3 clad sheet as given in QQ-A-250/5. | |
| | Inner Skin | 0.016 | 2024-T3 clad sheet as given in QQ-A-250/5. | |
| | Core | 0.500 | Aluminum honeycomb as given in BMS 4-4, Type 4-25, Class NPA. | |
| [9] | Inboard Main Flap, Leading Edge Skin | 0.080 | 2024-T42 clad sheet as given in QQ-A-250/5. | |

*[1] Note: T = Pre-manufactured thickness in inches.



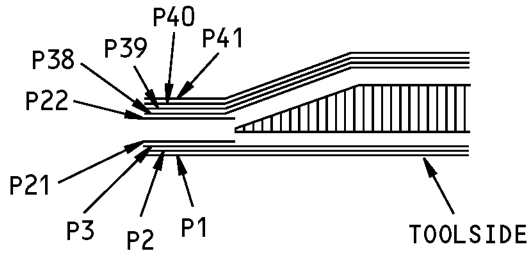

VIEW IS OF THE BAG SIDE (NON-AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTIONS

A

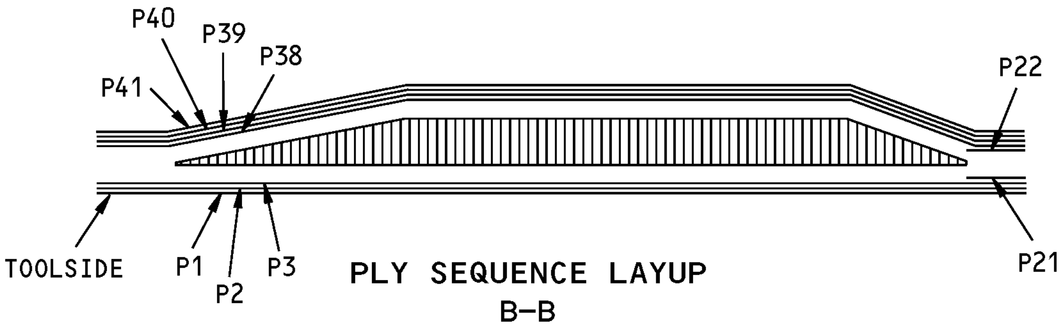
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, AND D-D FOR THE AREAS WHERE THE PLY SEQUENCE WILL BE IDENTIFIED.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

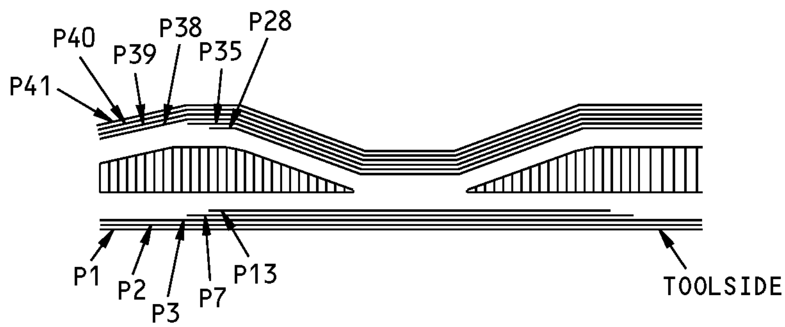
**Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]**
**Figure 3 (Sheet 1 of 2)**



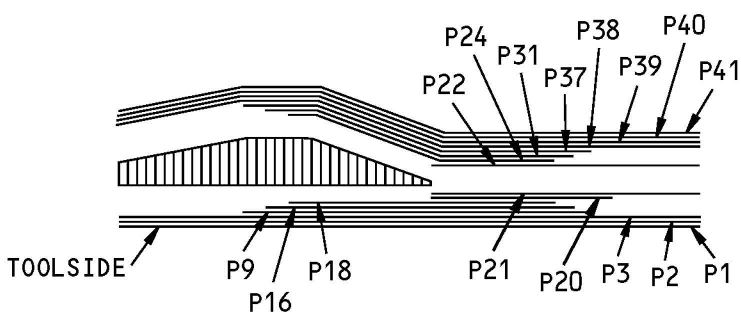

PLY SEQUENCE LAYUP
A-A




PLY SEQUENCE LAYUP
B-B




PLY SEQUENCE LAYUP
C-C

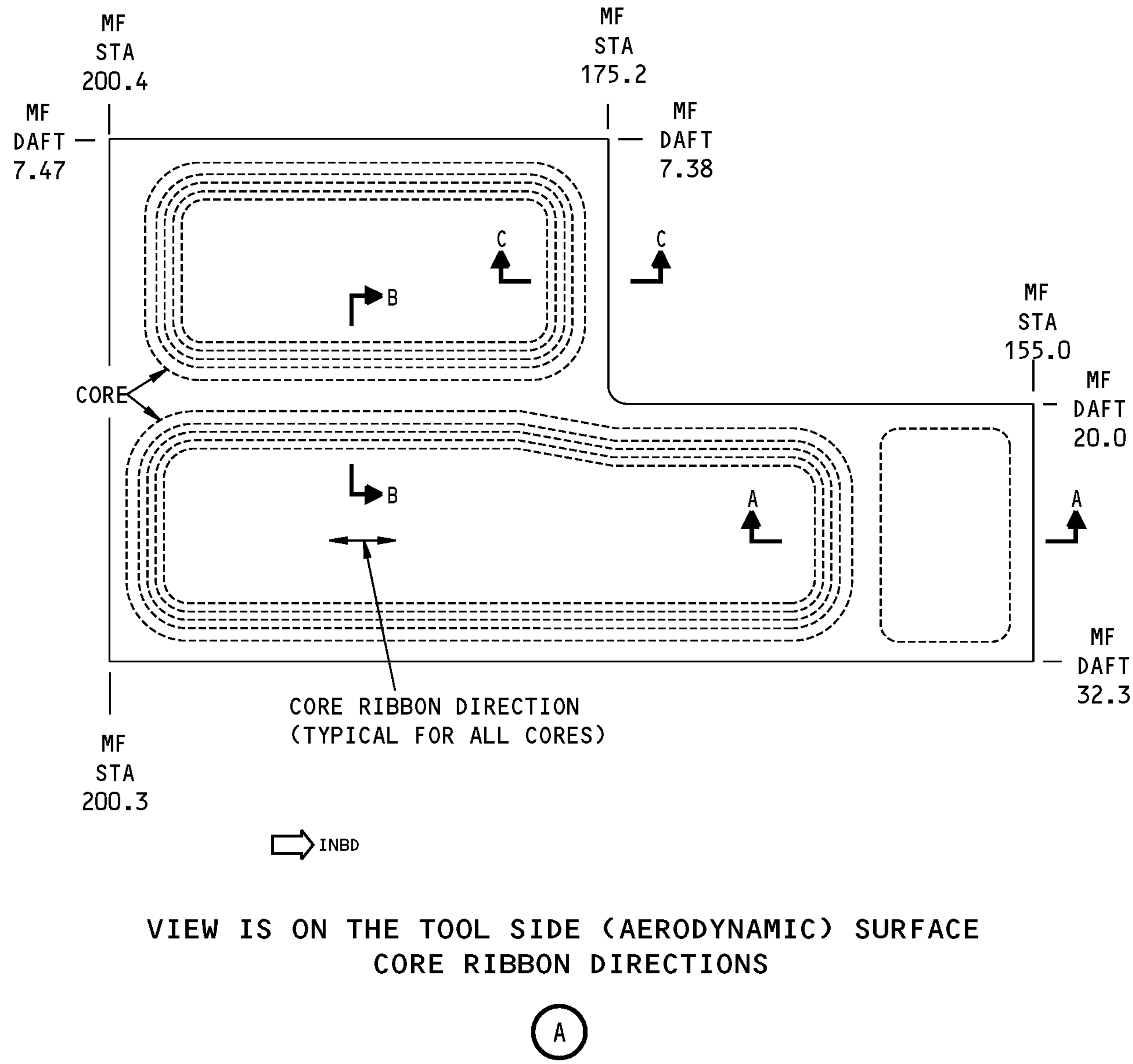



PLY SEQUENCE LAYUP
D-D

**Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]**
**Figure 3 (Sheet 2 of 2)**

**Table 3**:

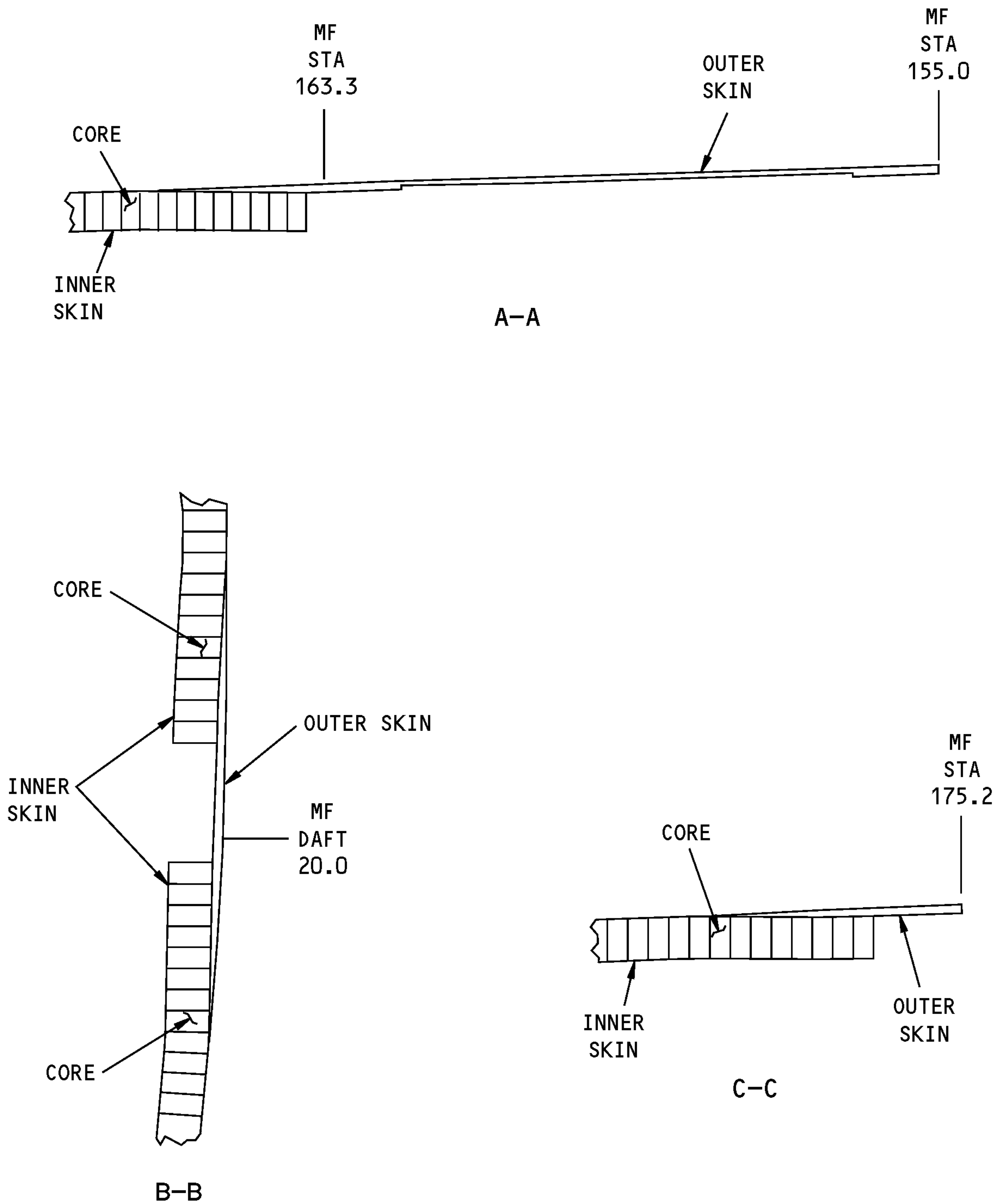
| PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [1] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 thru P40 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Style 1581, Grade B. |
| P41 | —— | 0.001 inch thick white bondable tedlar film |



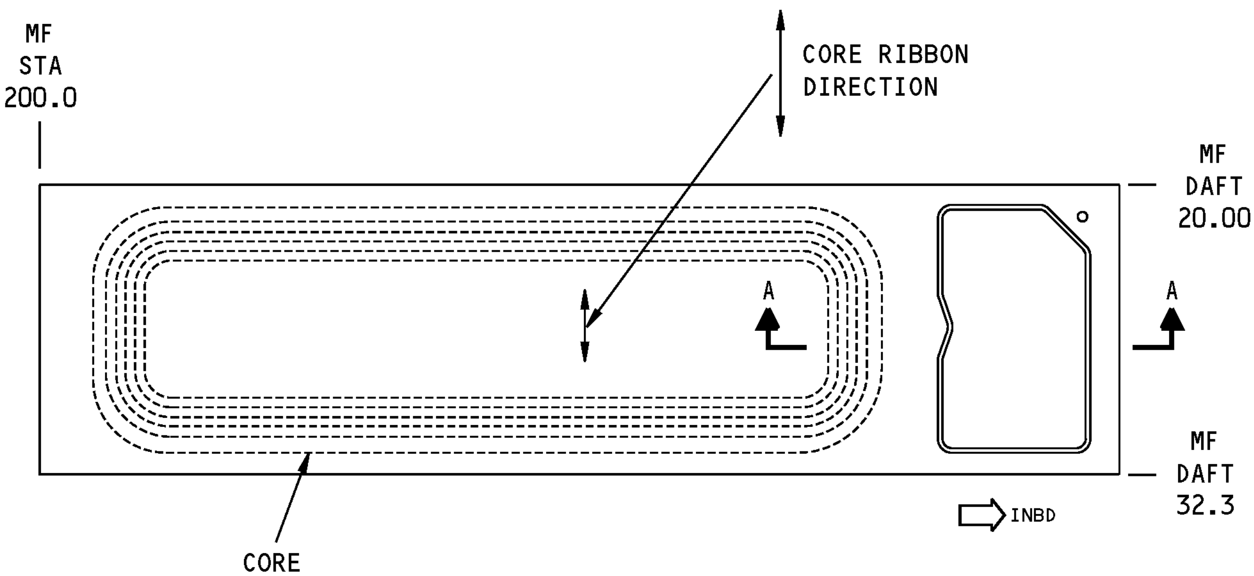

NOTES

- REFER TO DETAIL A FOR THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, AND C-C FOR THE BONDED AREAS.

**Core Ribbon Direction for Figure 2, Item [3]**
**Figure 4 (Sheet 1 of 2)**




**Core Ribbon Direction for Figure 2, Item [3]**
**Figure 4 (Sheet 2 of 2)**



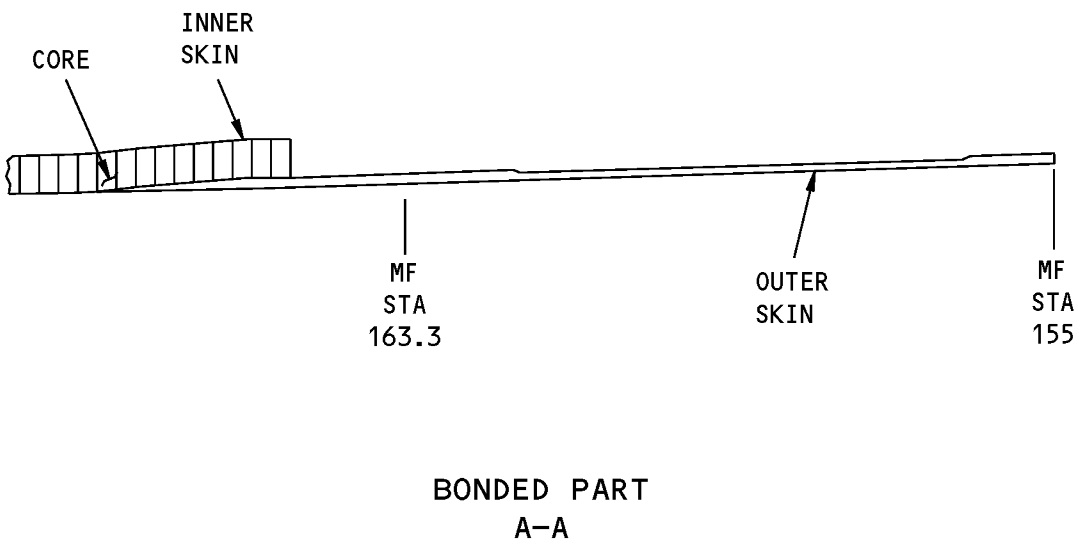

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
CORE RIBBON DIRECTIONS

A

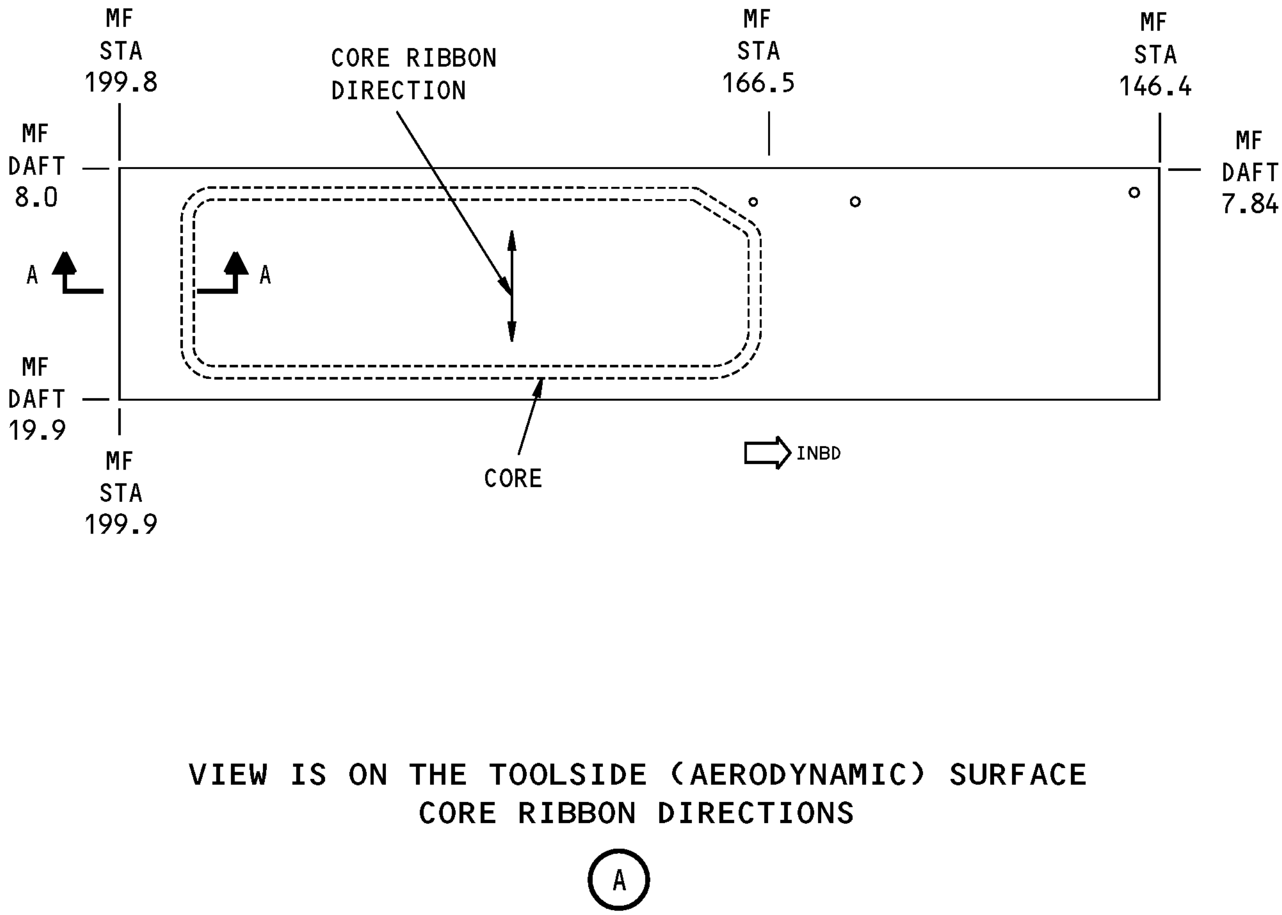
NOTES

- REFER TO DETAIL A FOR THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE BONDED AREAS.

**Core Ribbon Direction for Figure 2, Item [7]**
**Figure 5 (Sheet 1 of 2)**



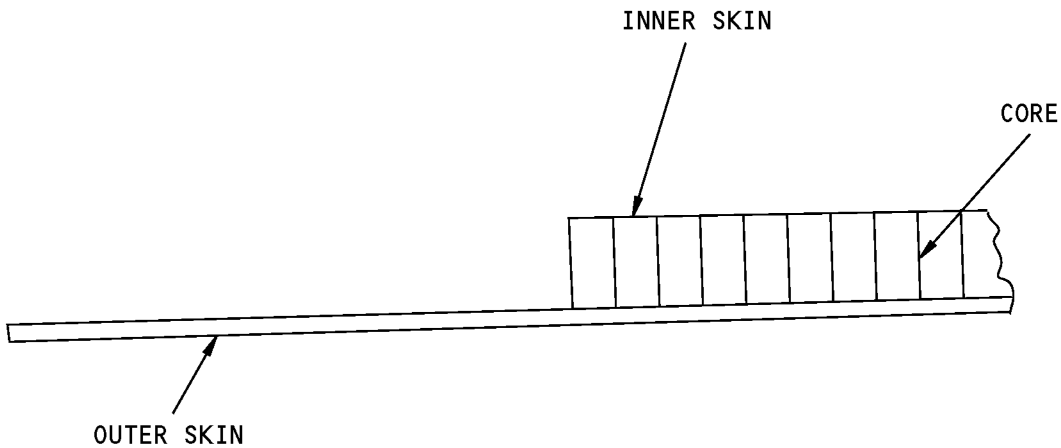

**Core Ribbon Direction for Figure 2, Item [7]**
**Figure 5 (Sheet 2 of 2)**




NOTES

- REFER TO DETAIL A FOR THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE BONDED AREAS.

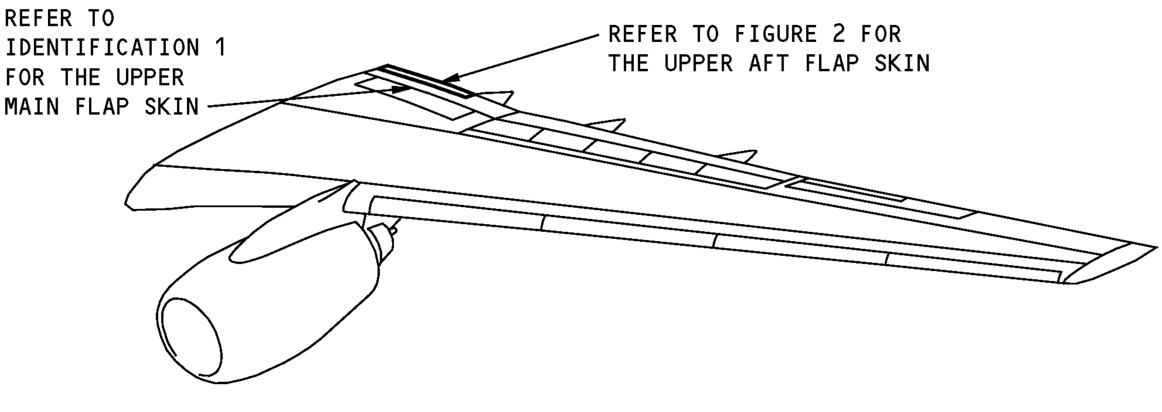
**Core Ribbon Direction for Figure 2, Item [8]**
**Figure 6 (Sheet 1 of 2)**

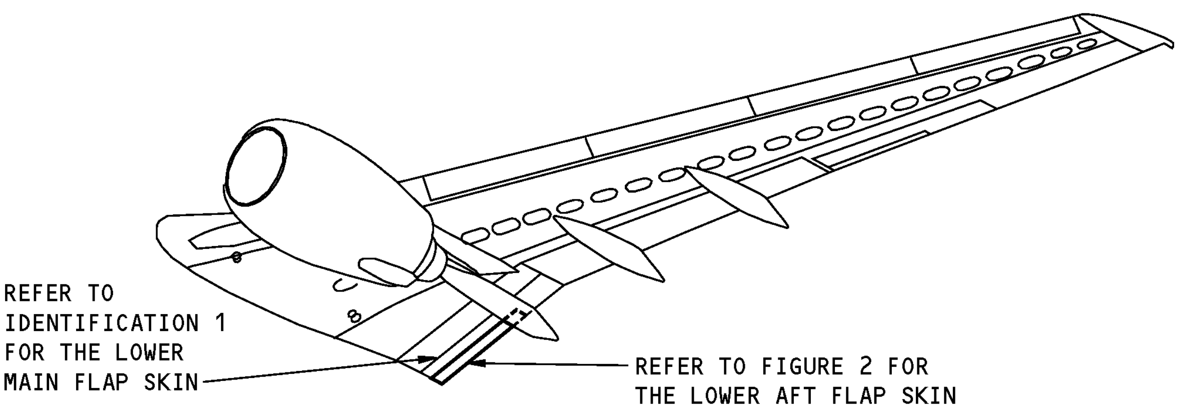



BONDED PART
A-A

**Core Ribbon Direction for Figure 2, Item [8]**
**Figure 6 (Sheet 2 of 2)**

## IDENTIFICATION 2 - WING TRAILING EDGE INBOARD AFT FLAP SKIN



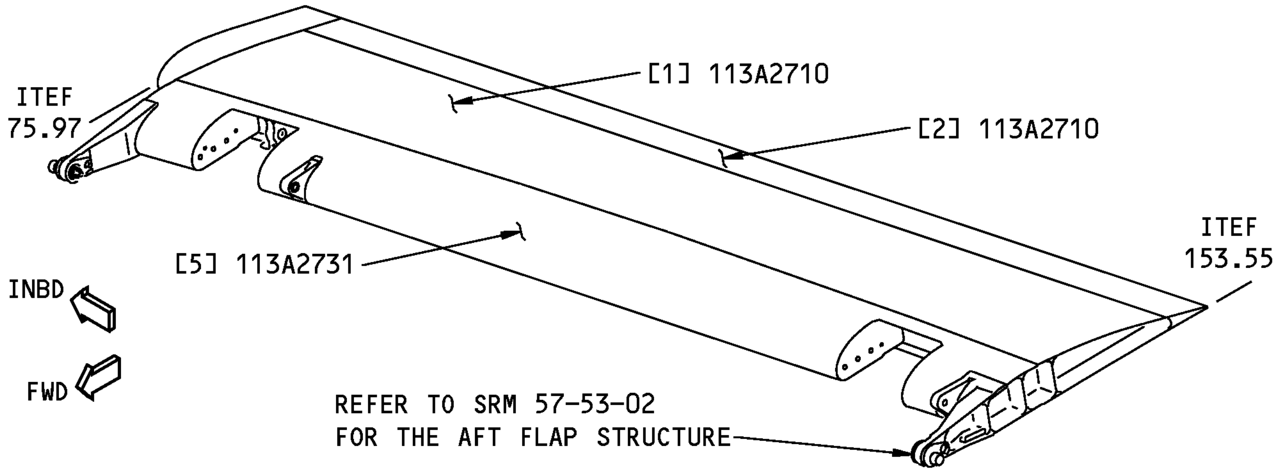




**Wing Trailing Edge - Inboard Aft Flap Skin Panel Location**
**Figure 1**

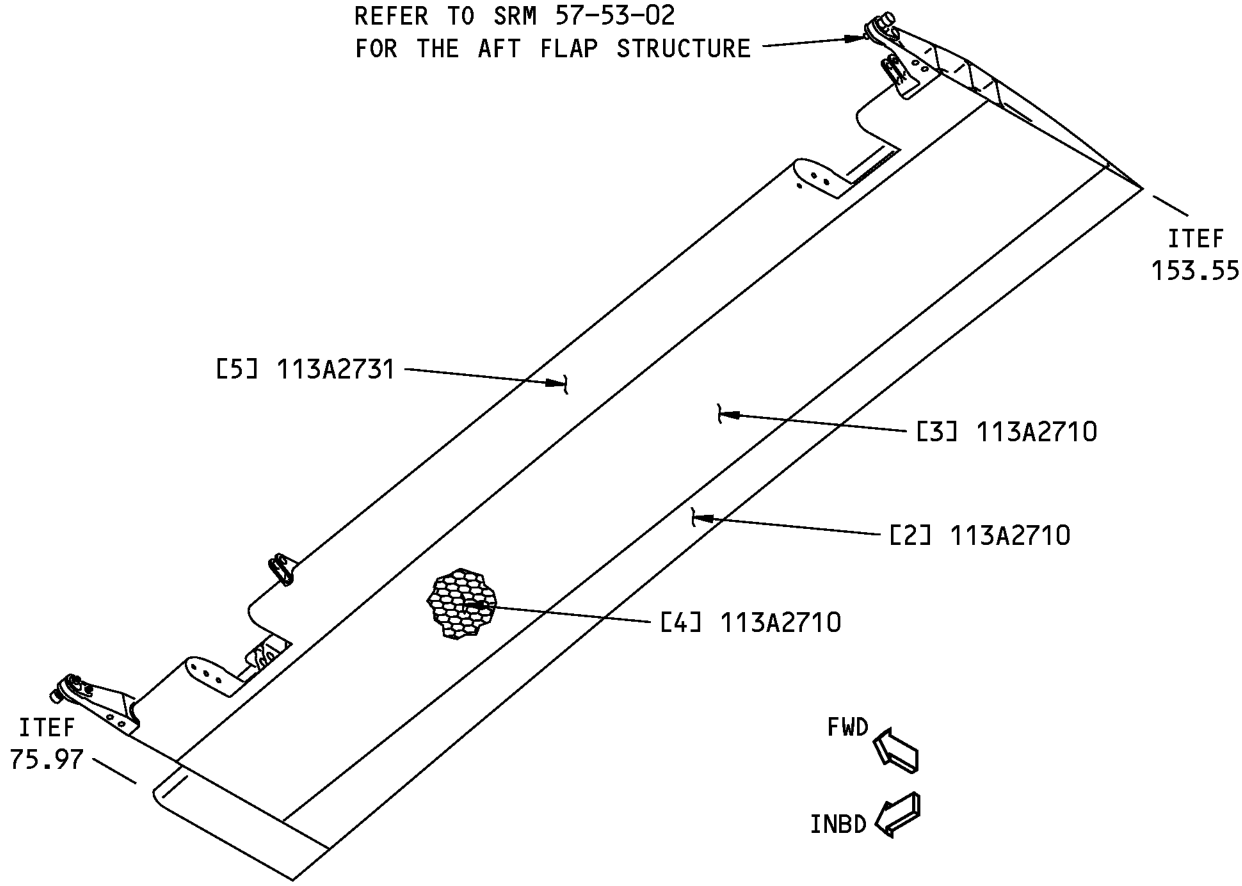
**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A2001 | Flap Installation - Inboard Flap |
| 113A2002 | Flap Assembly - Inboard Flap |
| 113A2100 | Main Flap Final Assembly - Inboard Flap |
| 113A2500 | Wedge Assembly - Bonded Part, Inboard Main Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A2700 | Aft Flap Assembly - Inboard Flap |
| 113A2710 | Bonded Assembly - Trailing Edge Wedge, Inboard Aft Flap |
| 113A2731 | Skin - Leading Edge, Inboard Aft Flap |




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

490522 S0000142583_V2

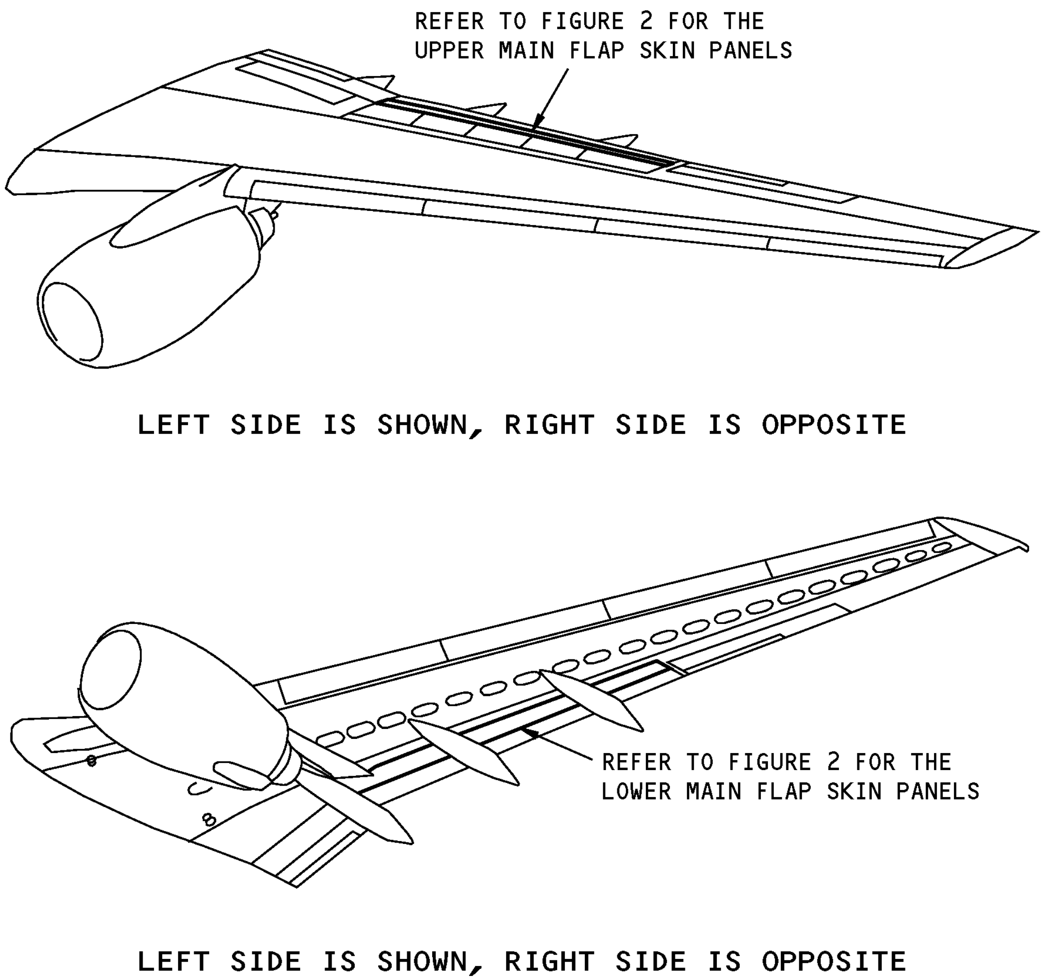
**Wing Trailing Edge - Inboard Aft Flap Skin Panel Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Upper Skin | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| [2] | Trailing Edge Filler | | Epoxy impregnated glass woven fiber as given in BMS 8-79, Class III, Grade B, Stye 1581 or Style 120 or Style 7781 | |
| [3] | Lower Skin | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | | | Glass fabric overlay as given in BMS 8-79, Class III, Style 7781 or BMS 9-3, Type H-2 as given in BAC 5337 | For cum line numbers 1131 and on |
| | | | (Optional) Glass fabric overlay as given in BMS 8-79, Class III, Style 7781 or BMS 9-3, Type H-2 as given in BAC 5337 | For cum line numbers 1 thru 1130 |
| [4] | Honeycomb Core | | Aluminum honeycomb core as given in BMS 4-4, Type 4-25, Class NPA, Form B, Grade I | |
| [5] | Leading Edge SKin | 0.071 (1.80) | 2024-T42 clad sheet as given in QQ-A-250/5 | |

*[1] Note: T = Pre-manufactured thickness in inches.

## IDENTIFICATION 3 - WING TRAILING EDGE OUTBOARD MAIN FLAP SKIN

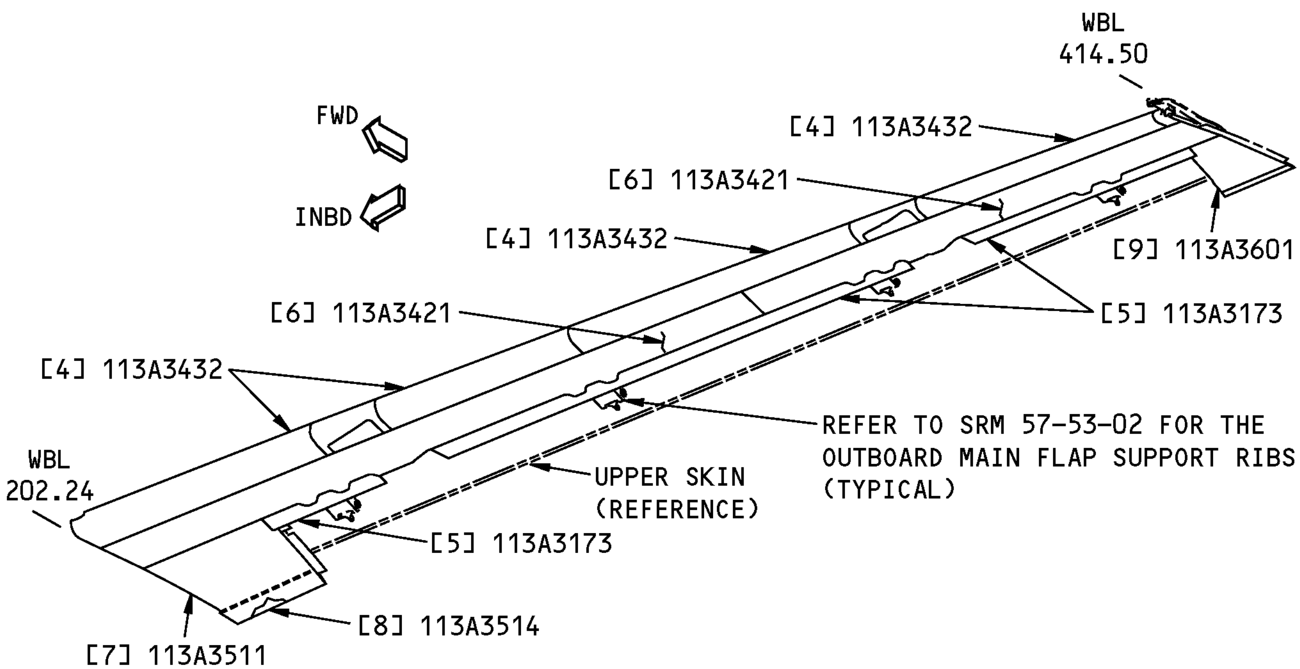



**Wing Trailing Edge - Outboard Main Flap Skin Panel Location**
**Figure 1**

**Table 1**:

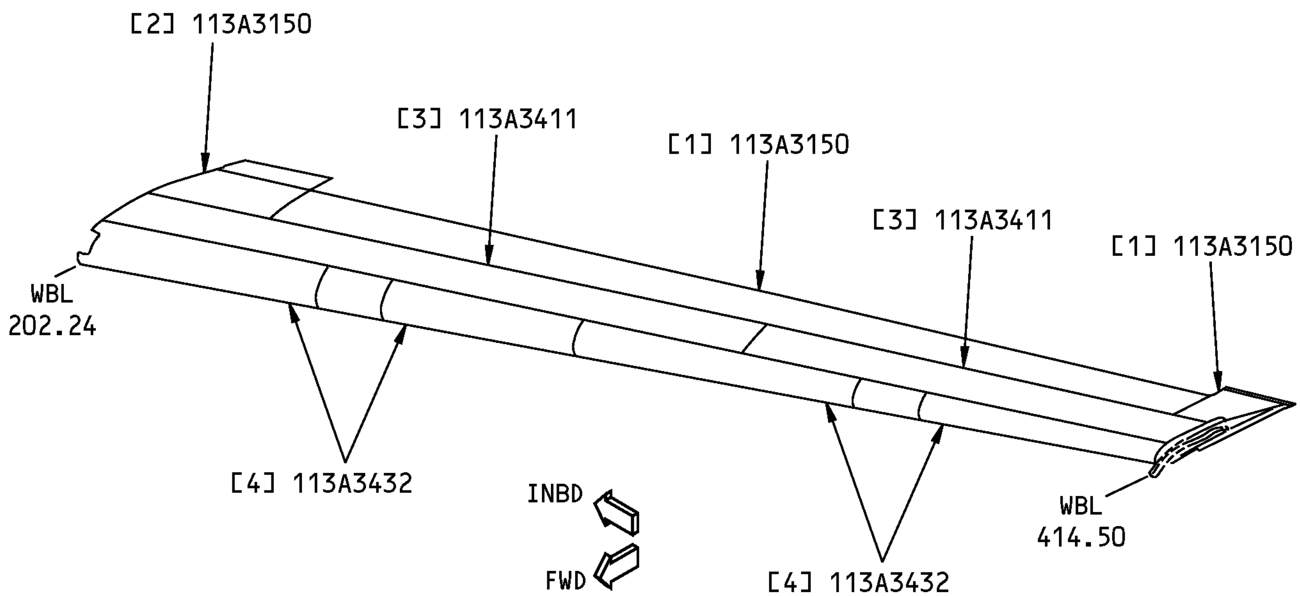
| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A3001 | Flap Installation - Outboard Trailing Edge Flap |
| 113A3002 | Flap Assembly - Outboard Trailing Edge Flap |
| 113A3100 | Main Flap Assembly - Outboard Trailing Edge Flap |
| 113A3150 | Upper Cove Panel - Outboard Main Trailing Edge Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A3173 | Lower Cove Panel - Outboard Main Trailing Edge Flap |
| 113A3411 | Skin Panel - Upper Forward, Outboard Main Trailing Edge Flap |
| 113A3421 | Skin Panel - Lower Forward, Outboard Main Trailing Edge Flap |
| 113A3432 | Skin Panel - Leading Edge, Outboard Main Trailing Edge Flap |
| 113A3511 | Inboard Trailing Edge Wedge, Outboard Main Trailing Edge Flap |
| 113A3514 | Skin Panel - Inboard Lower, Aft, Outboard Main Trailing Edge Flap |
| 113A3601 | Skin Panel - Outboard Lower, Aft, Outboard Main Trailing Edge Flap |




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD MAIN FLAP LOWER SKIN PANEL




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

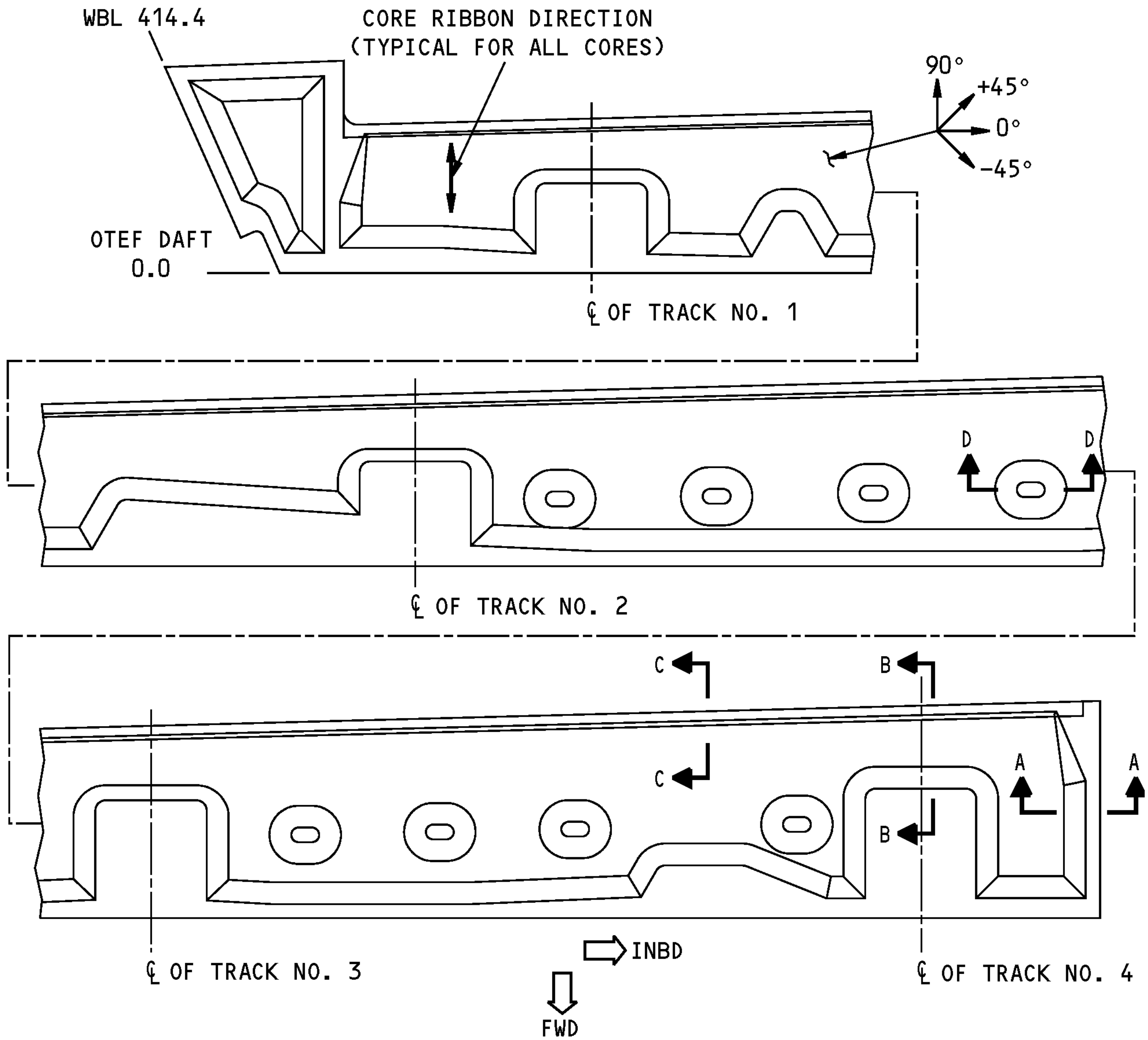
OUTBOARD MAIN FLAP UPPER SKIN PANEL

**Wing Trailing Edge - Outboard Main Flap Skin Panel Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Upper Outboard Trailing Edge Cove Panel - Bonded Part | | Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class 3, Grade B, Style 1581 | |
| | Skin | | Refer to Figure 3 | |
| | Core (outboard) | 0.500 | Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction. | |
| | Core (inboard) | 0.500 | Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction. | |
| [2] | Upper Inboard Trailing Edge Cove Panel - Bonded Part | | Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class 3, Grade B, Style 1581 | |
| | Skin | | Refer to Figure 4 | |
| | Core (outboard) | 0.500 | Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction. | |
| | Core (inboard) | 0.500 | Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction. | |
| [3] | Upper Forward Skin | 0.100 | 7075-T62 bare sheet as given in QQ-A-250/12. | |
| [4] | Leading Edge Skin | 0.080 | 2024-T42 clad sheet as given in QQ-A-250/5. | |
| [5] | Lower Trailing Edge Cove Skin | 0.080 | 2024-T3 bare sheet as given in QQ-A-250/4. | |
| [6] | Lower Forward Skin | 0.080 | 2024-T3 bare sheet as given in QQ-A-250/4. | |
| [7] | Inboard Trailing Edge Wedge | 2.500 | 7050-T7451 plate as given in AMS 4050. | |
| [8] | Lower Inboard Aft Skin | 0.080 | 2024-T3 bare sheet as given in QQ-A-250/4. | |
| [9] | Lower Outboard Aft Skin | 0.080 | 2024-T3 bare sheet as given in QQ-A-250/4. | |

*[1] Note: T = Pre-manufactured thickness in inches.




VIEW IS OF THE BAG SIDE (NON-AERODYNAMIC) SURFACE

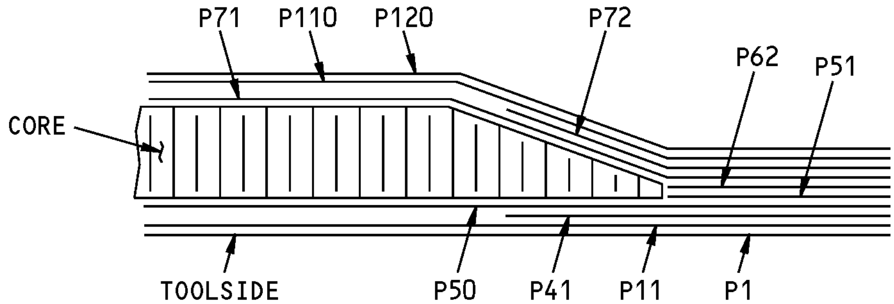
PLY LAYUP AND CORE RIBBON DIRECTIONS

A

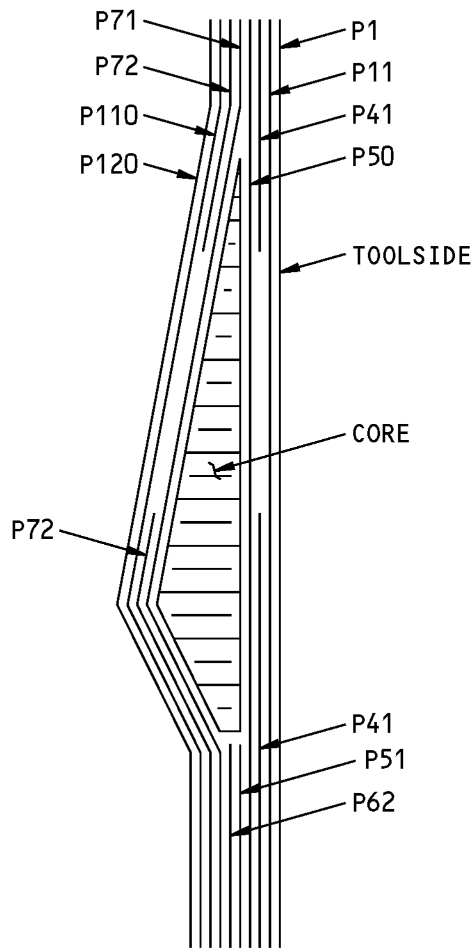
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, AND D-D FOR THE AREAS WHERE THE PLY SEQUENCE WILL BE IDENTIFIED.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]**
**Figure 3 (Sheet 1 of 3)**

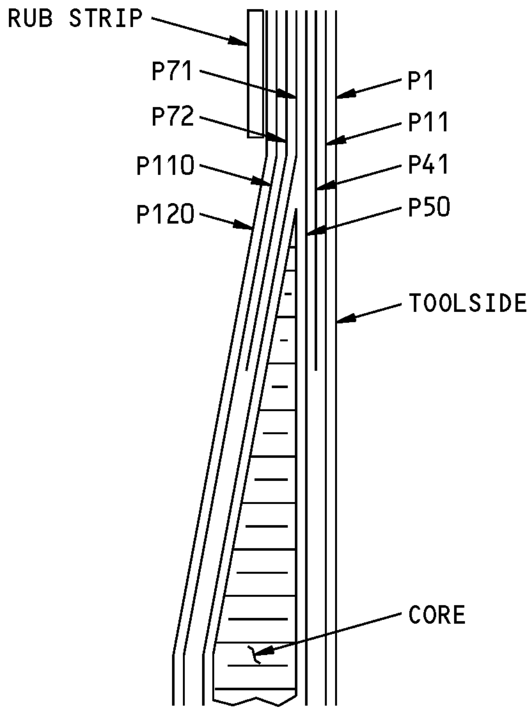



PLY SEQUENCE LAYUP
A-A

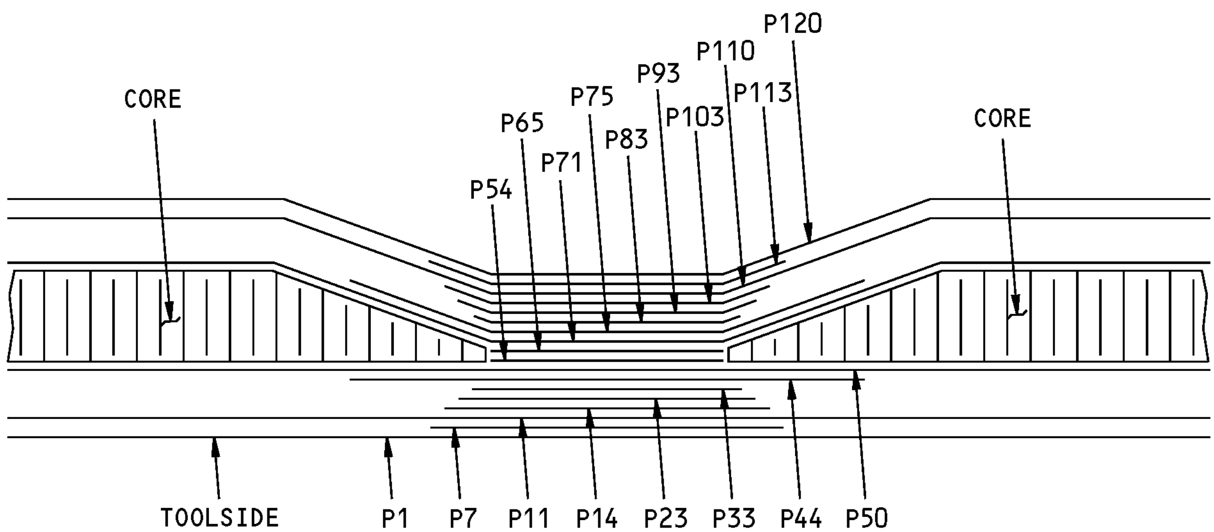



PLY SEQUENCE LAYUP
B-B

**Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]**
**Figure 3 (Sheet 2 of 3)**




PLY SEQUENCE LAYUP
C-C

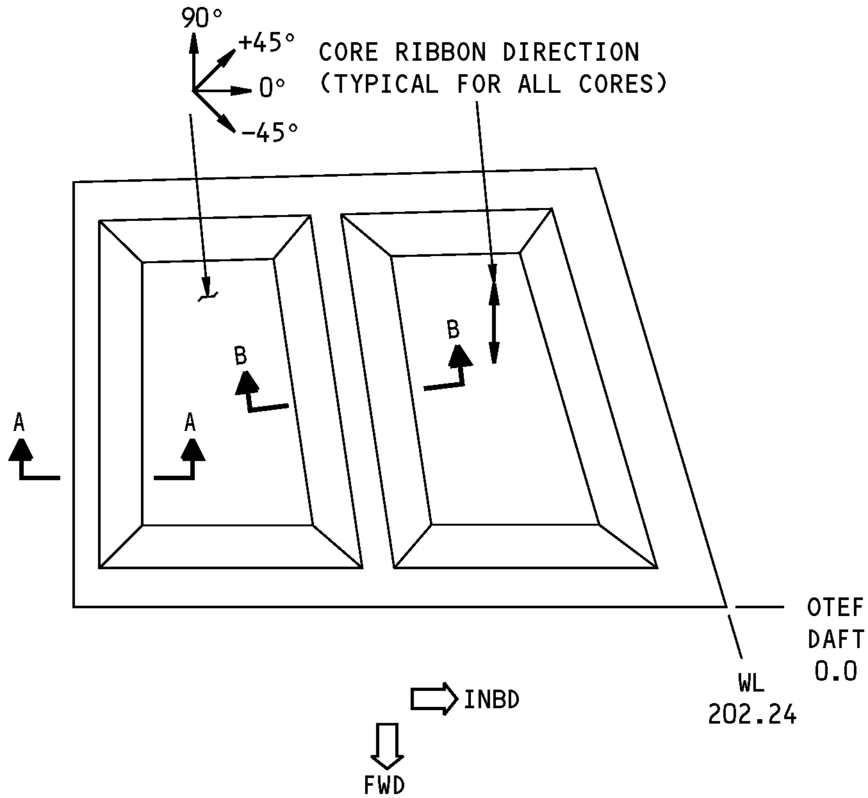



PLY SEQUENCE LAYUP
D-D

**Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]**
**Figure 3 (Sheet 3 of 3)**

**Table 3**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [1] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P5 thru P10, P21 thru P30, P50 thru P59, P62 thru P71, P91 thru P100, P111 thru P116, P120. | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. |
| P2 thru P4, P11 thru P20, P31 thru P49, P60 thru P61, P72 thru P90, P101 thru P110, P117 thru P119. | + or – 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. |




VIEW IS OF THE BAG SIDE (NON-AERODYNAMIC) SURFACE

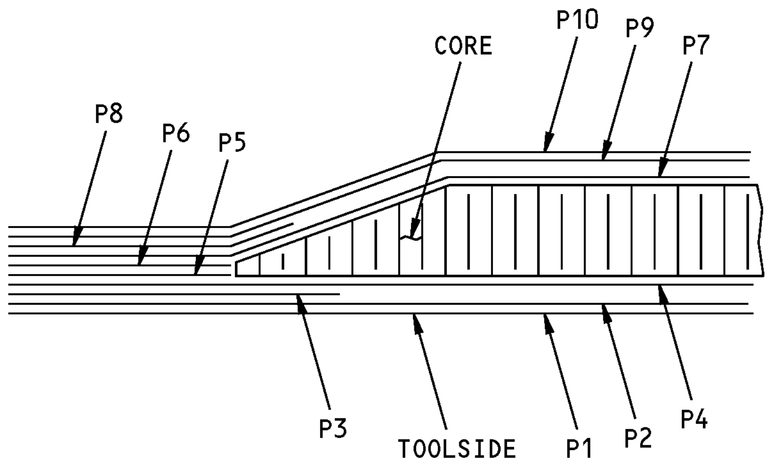
PLY LAYUP AND CORE RIBBON DIRECTIONS

A

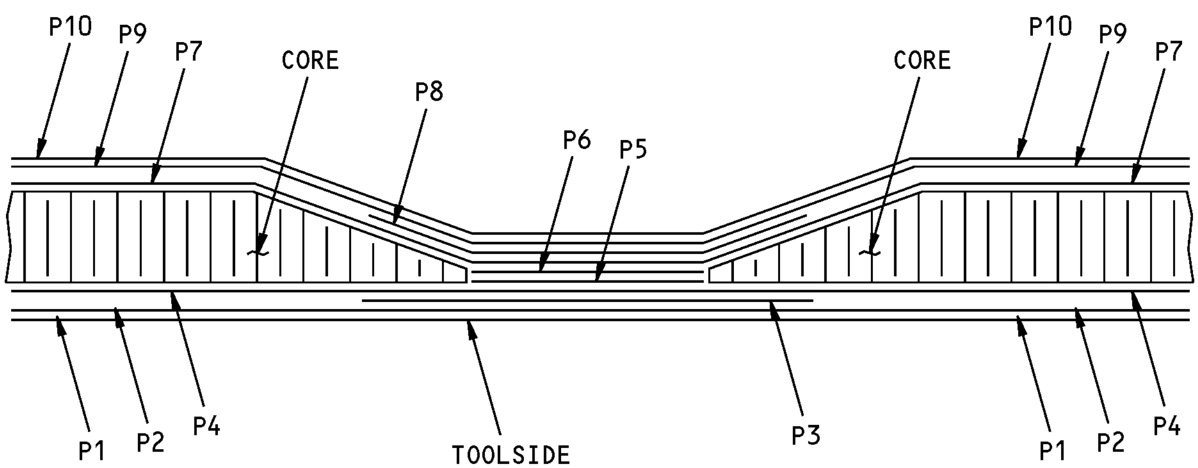
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE AREAS WHERE THE PLY SEQUENCE WILL BE IDENTIFIED.
- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [2]**
**Figure 4 (Sheet 1 of 2)**




PLY SEQUENCE LAYUP
A-A



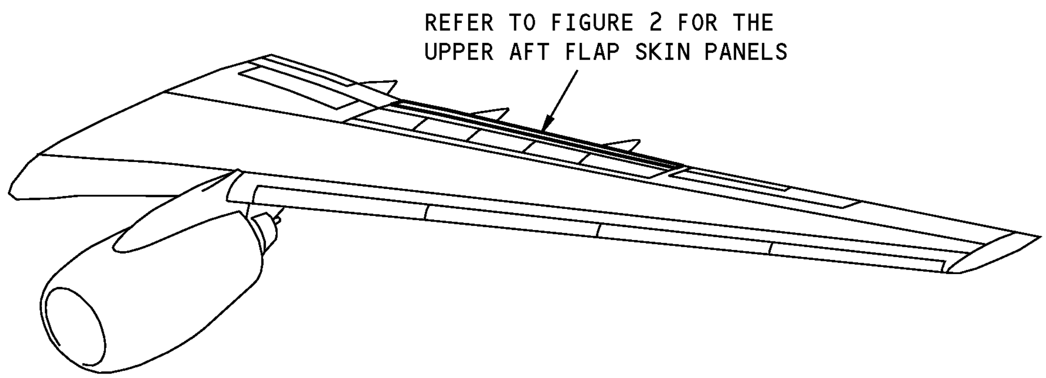

PLY SEQUENCE LAYUP
B-B

**Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [2]**
**Figure 4 (Sheet 2 of 2)**

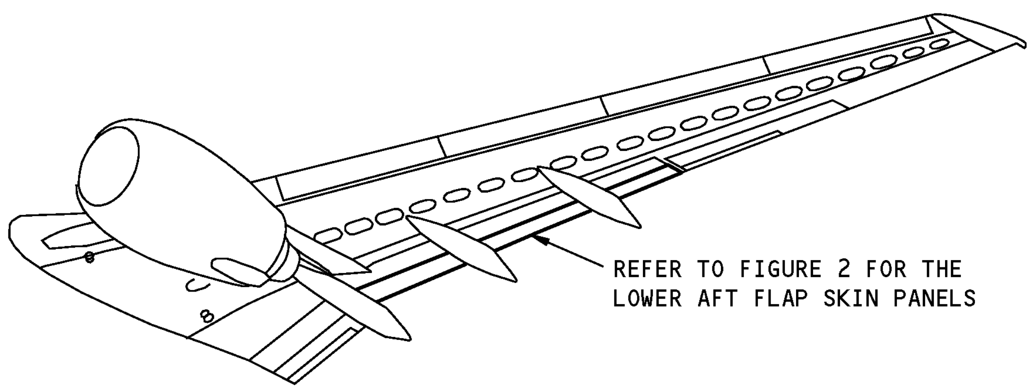
Table 4:

| PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [2] | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1, P4 thru P7, P10. | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. |
| P2 thru P3, P8 thru P9. | + or – 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. |

## IDENTIFICATION 4 - WING TRAILING EDGE OUTBOARD AFT FLAP SKIN




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE



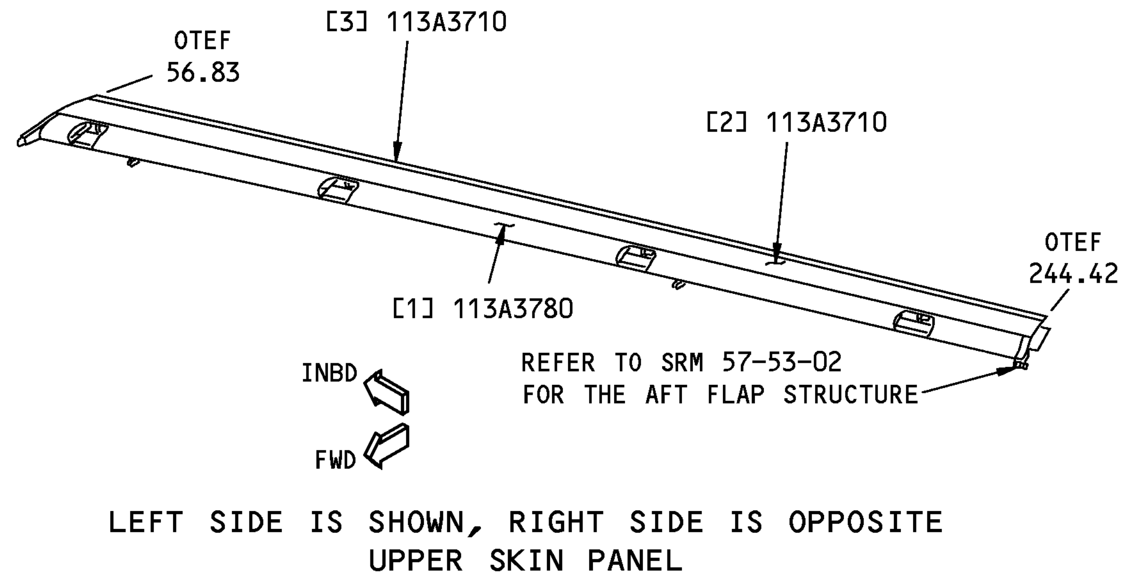

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

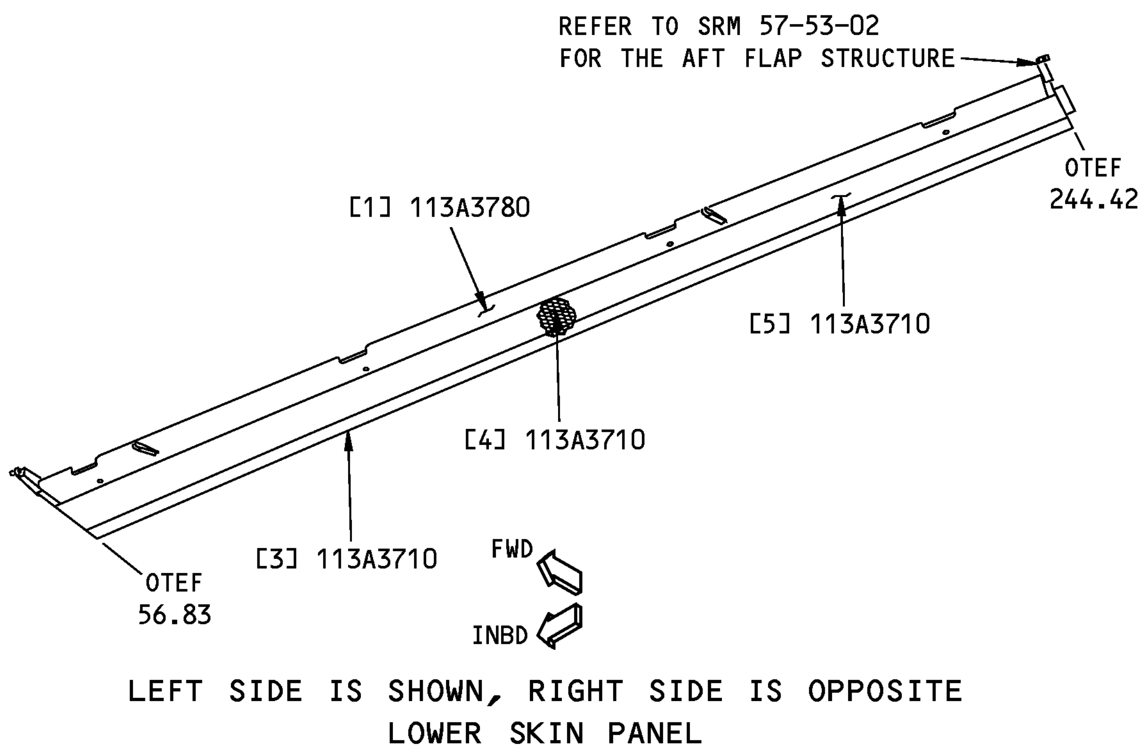
**Wing Trailing Edge - Outboard Aft Flap Skin Panel Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A3001 | Flap Installation - Outboard Trailing Edge Flap |
| 113A3002 | Flap Assembly - Outboard Trailing Edge Flap |
| 113A3700 | Aft Flap Assembly - Outboard Trailing Edge Flap |
| 113A3710 | Wedge Assembly - Bonded Part, Outboard Trailing Edge Aft Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A3780 | Skin Panel - Leading Edge, Outboard Trailing Edge Aft Flap |




490524 S0000144415_V2

**Wing Trailing Edge - Outboard Aft Flap Skin Panel Identification**
**Figure 2**

**Table 2**:

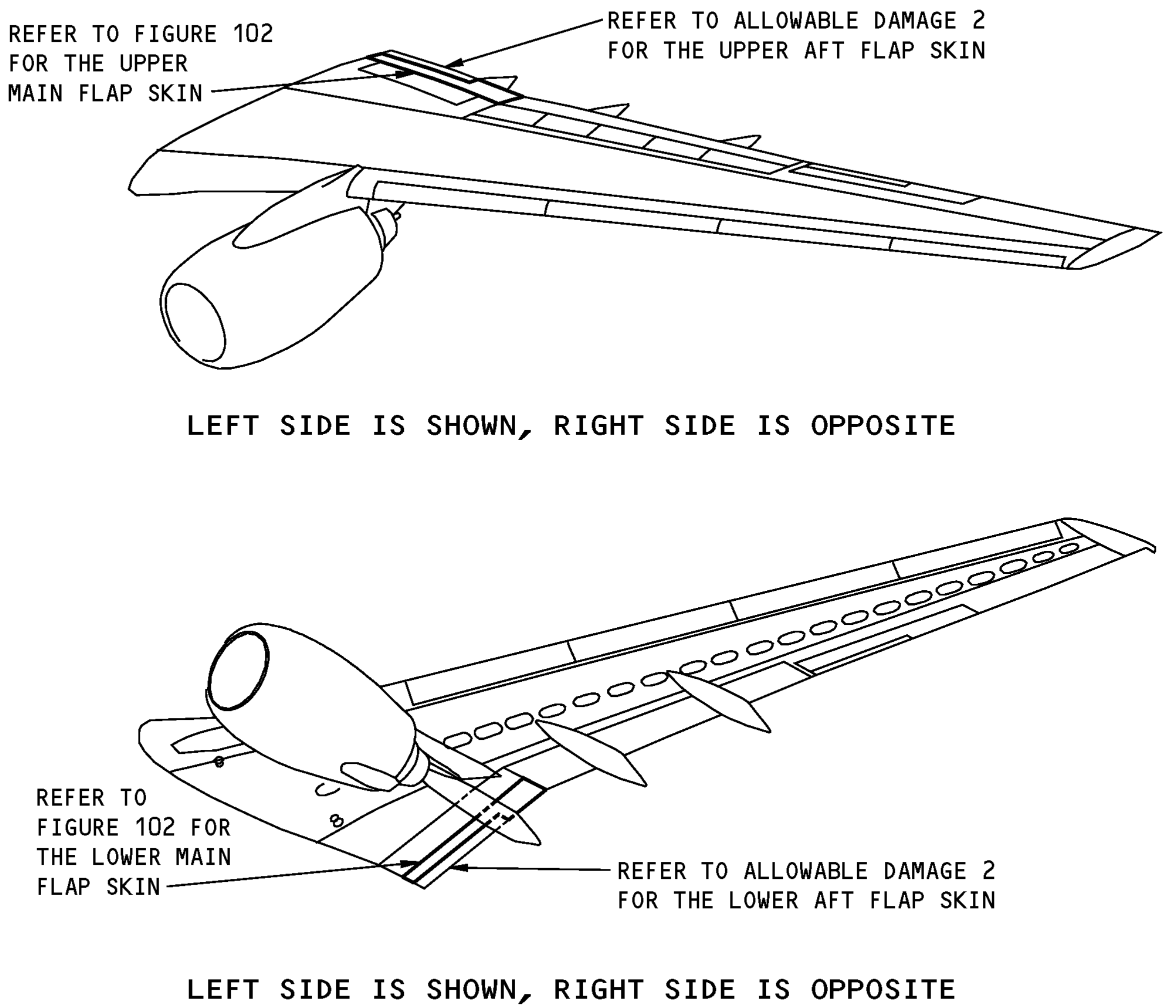
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Skin - Leading Edge | 0.071 (1.80) | 2024-T42 clad sheet as given in QQ-A-250/5 | |
| [2] | Skin - Upper | 0.016 (0.41) | 7075-T6 bare sheet as given in QQ-A-250/12 | |
| [3] | Fillet | | Glass Fiber Reinforced Plastic (GFRP) Pre-preg as given in BMS 8-79, Style 1581 or 120 | |
| [4] | Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 4-25 NPA | |
| [5] | Skin - Lower | 0.020 (0.51) | 7075-T6 bare sheet as given in QQ-A-250/12 | |

*[1] Note: T = Pre-manufactured thickness in inches.

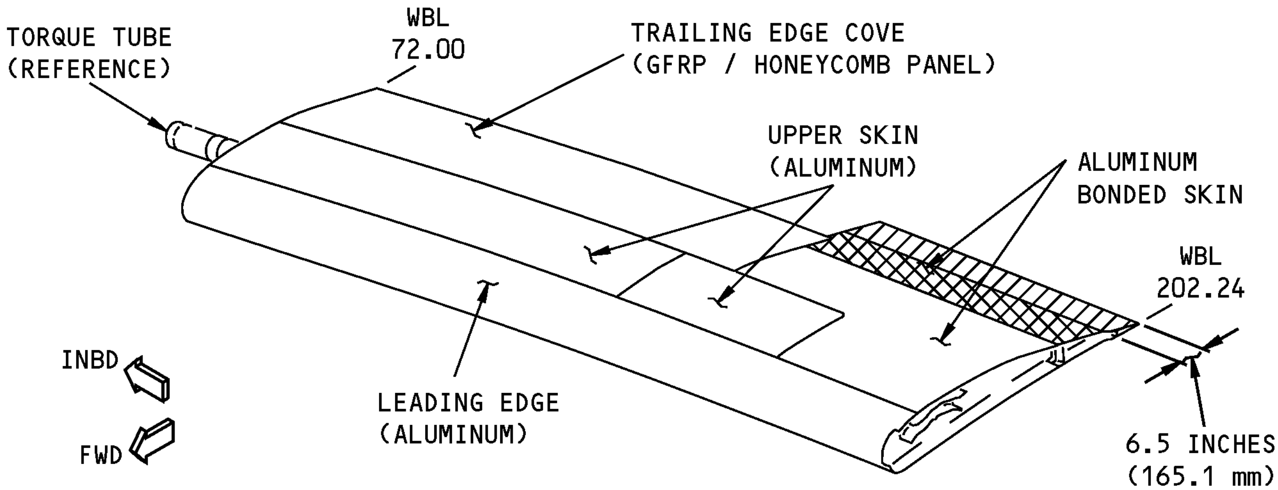
## ALLOWABLE DAMAGE 1 - WING INBOARD TRAILING EDGE MAIN FLAP

### 1. Applicability

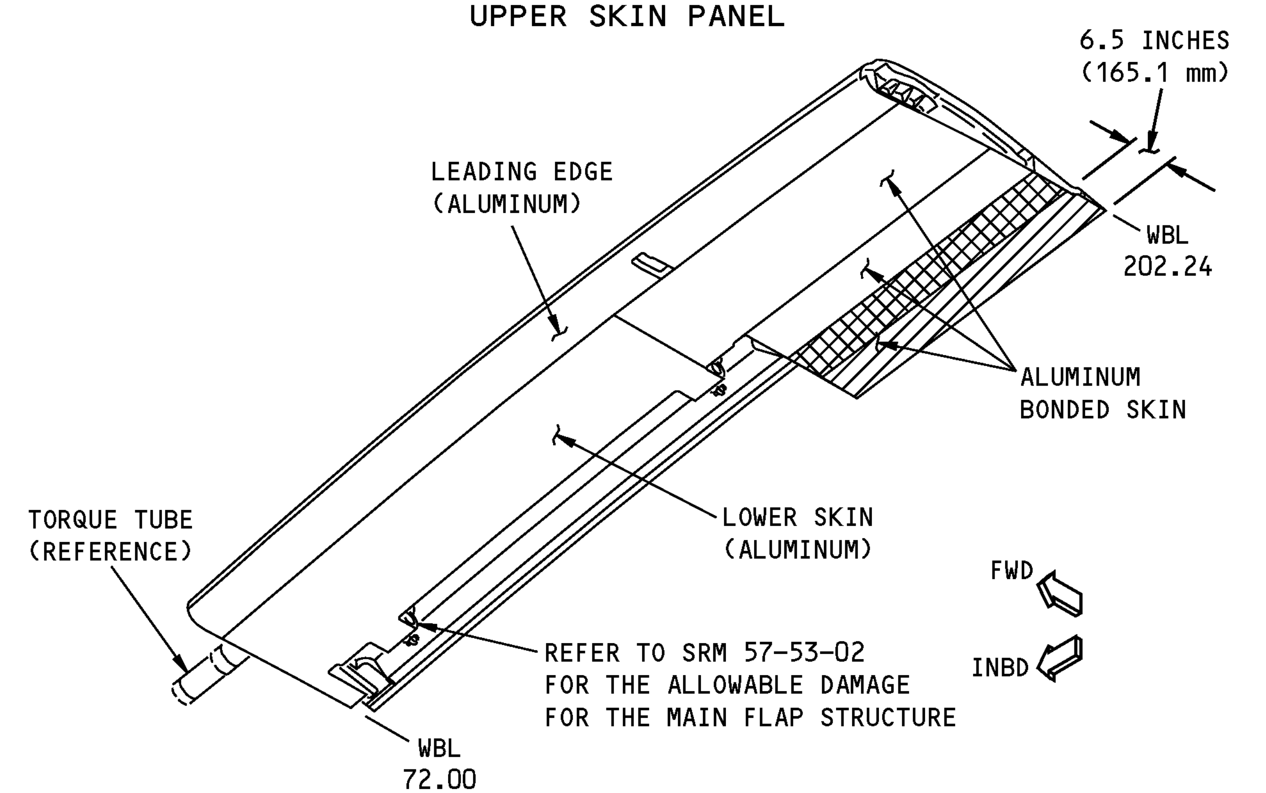
A. This subject gives the allowable damage limits for the wing inboard trailing edge main flap as shown in Wing Inboard Trailing Edge - Main Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.




**Wing Inboard Trailing Edge - Main Flap Skin Panel Location**
**Figure 101**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

ZONE 1 - DELAMINATION LIMITS

ZONE 2 - DELAMINATION LIMITS

**Wing Inboard Trailing Edge - Main Flap Skin Panel Allowable Damage**
**Figure 102**

**2. General**

A. Do the steps that follow if you have damage to the aluminum skins:

(1) Remove the damaged material. Refer to 51-10-02 for the procedures.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(4) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

B. Do the steps that follow if you find damage on the trailing edge cove made of Glass Fiber Reinforced Plastic (GFRP) on the upper skin panel.

(1) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits in Paragraph 4./ALLOWABLE DAMAGE 1

(a) Make a temporary seal.

1) Apply aluminum foil tape (speed tape).

2) Keep a record of the location.

3) Make sure the tape is in satisfactory condition at each 400 flight hour interval or more frequently.

4) Seal the damage permanently no later than 5000 flight hours.

(b) Make a permanent seal.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to AMM PAGEBLOCK 51-21-99/701.

(2) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

(a) Use a vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Repair the damage no later than 400 flight hours.

**3. References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |

(Continued)

| Reference | Title |
|---|---|
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| 51-70-10 | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |

4. **Allowable Damage Limits**

A. Leading Edge, Upper, Lower, and Bonded Skins (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than 0.125 inch and less than or equal to 0.250 inch.

b) The distance between two dents or between a dent and other damage is less than 0.50D (D = the larger dimension of the two damage areas.)

2) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

(4) Delamination outside of the Zone 1 and 2 locations is not permitted.

(5) Zone 1 - Upper and Lower Skin Delamination

(a) The damage is permitted if:

1) It is a maximum of 2.5 inches (63.5 mm) in diameter.

2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

3) It is a minimum of 1.5 inches (38.1 mm) away from the outboard and inboard edge of the facesheet.

4) You apply a temporary seal with aluminum foil tape (speed tape) and do the steps that follow:

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(6) Zone 2 - Upper and Lower Skin Delamination

(a) The damage is permitted if:

1) It is a maximum of 1.0 inch (25.4 mm) in diameter.

2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

3) It is a minimum of 1.0 inch (25.4 mm) aft of the spar web.

4) You apply a temporary seal with aluminum foil tape (speed tape) and do the steps that follow:

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(7) Holes and Punctures outside of the Zone 1 and 2 locations are not permitted.

(8) Zones 1 and 2 - Upper and Lower Skin Holes and Punctures

(a) The damage is permitted if:

1) It is a maximum of 0.50 inch (12.7 mm) in diameter.

2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

3) It is a minimum of 1.0 inch (25.4 mm) aft of the spar web.

4) You apply a temporary seal with aluminum foil tape (speed tape) and do the steps that follow:

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

B. Trailing Edge Cove (Upper Skin)

**NOTE**: For edge damage only, refer to Damage Definitions, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth.

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply.

(b) A maximum of 3.0 inches in length.

(c) A maximum of 0.25 inch in width.

(d) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(3) Dents are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

**NOTE**: Use the limits for holes and punctures if there is glass fiber damage.

(b) The damage is a minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

(b) A maximum of one facesheet and the core in depth.

(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A.

(5) Delaminations are permitted as shown in Figure 105 (Sheet 1), Detail A if it is:

(a) A maximum of 0.50 inch in length.

(b) A maximum of 0.50 inch in width.

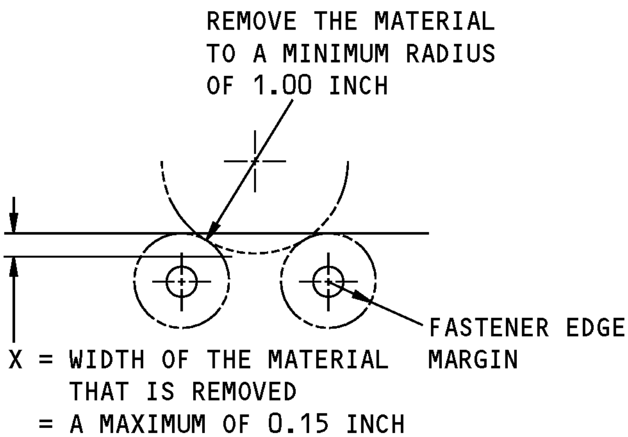
(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth.

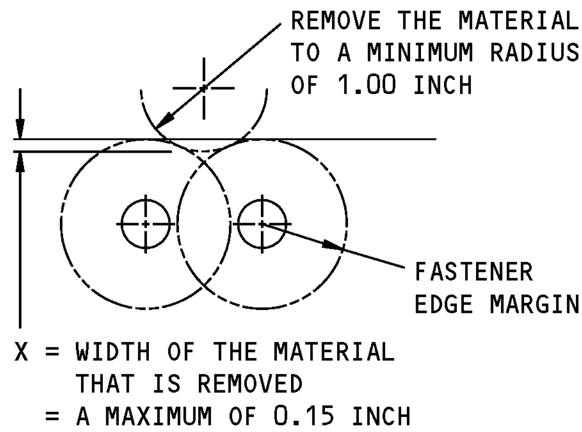
(b) A maximum of 0.25 inch in width.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail B .



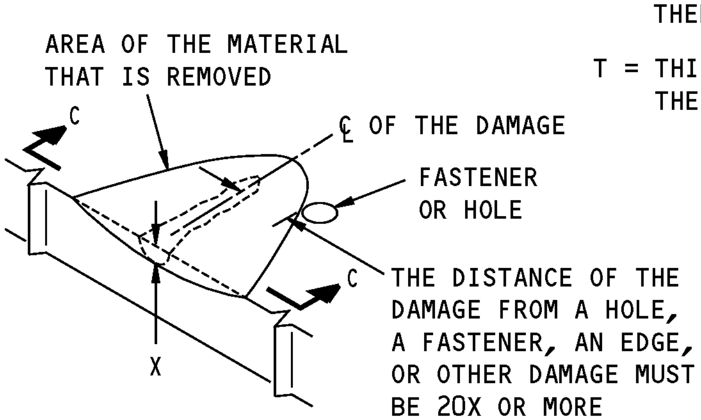

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A



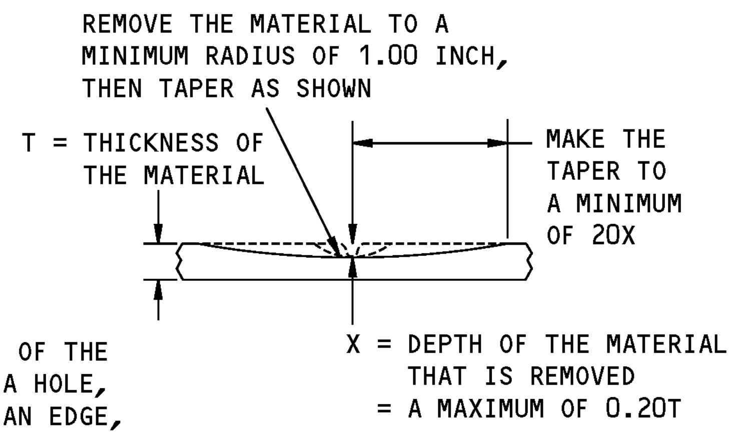

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B



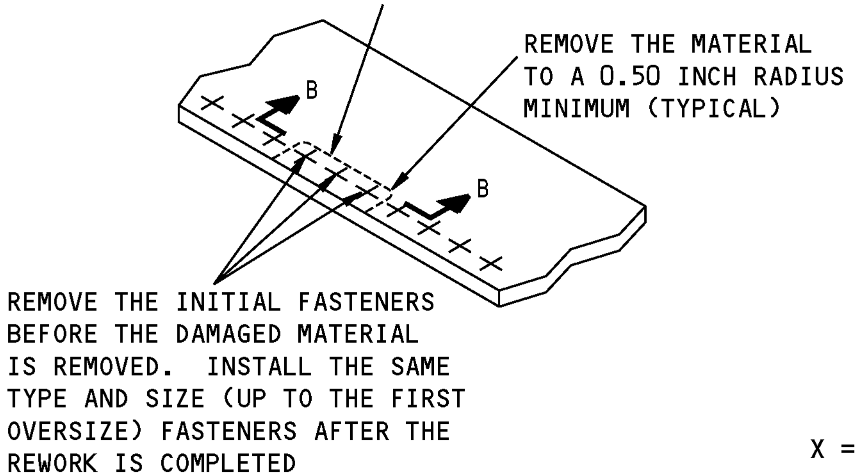

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C



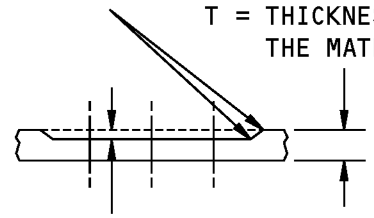

A-A

**Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage**
**Figure 103 (Sheet 1 of 3)**

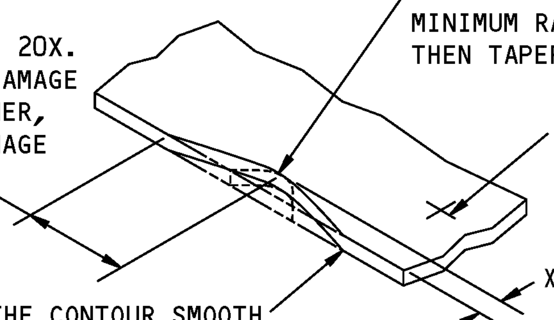



REMOVAL OF CORROSION AROUND THE FASTENERS

D




B-B

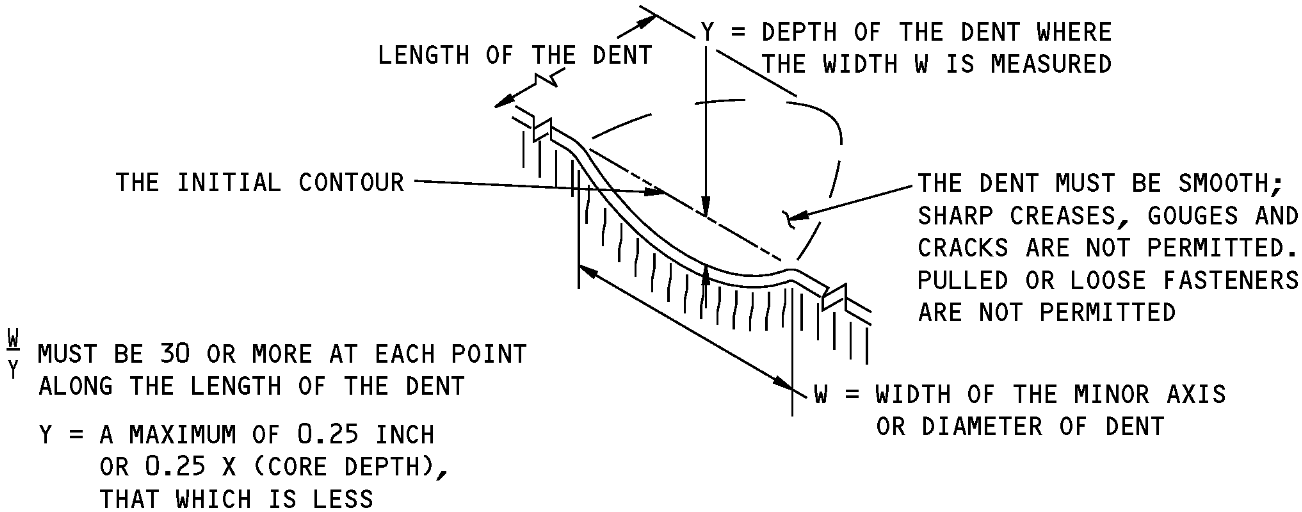



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.15 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

E

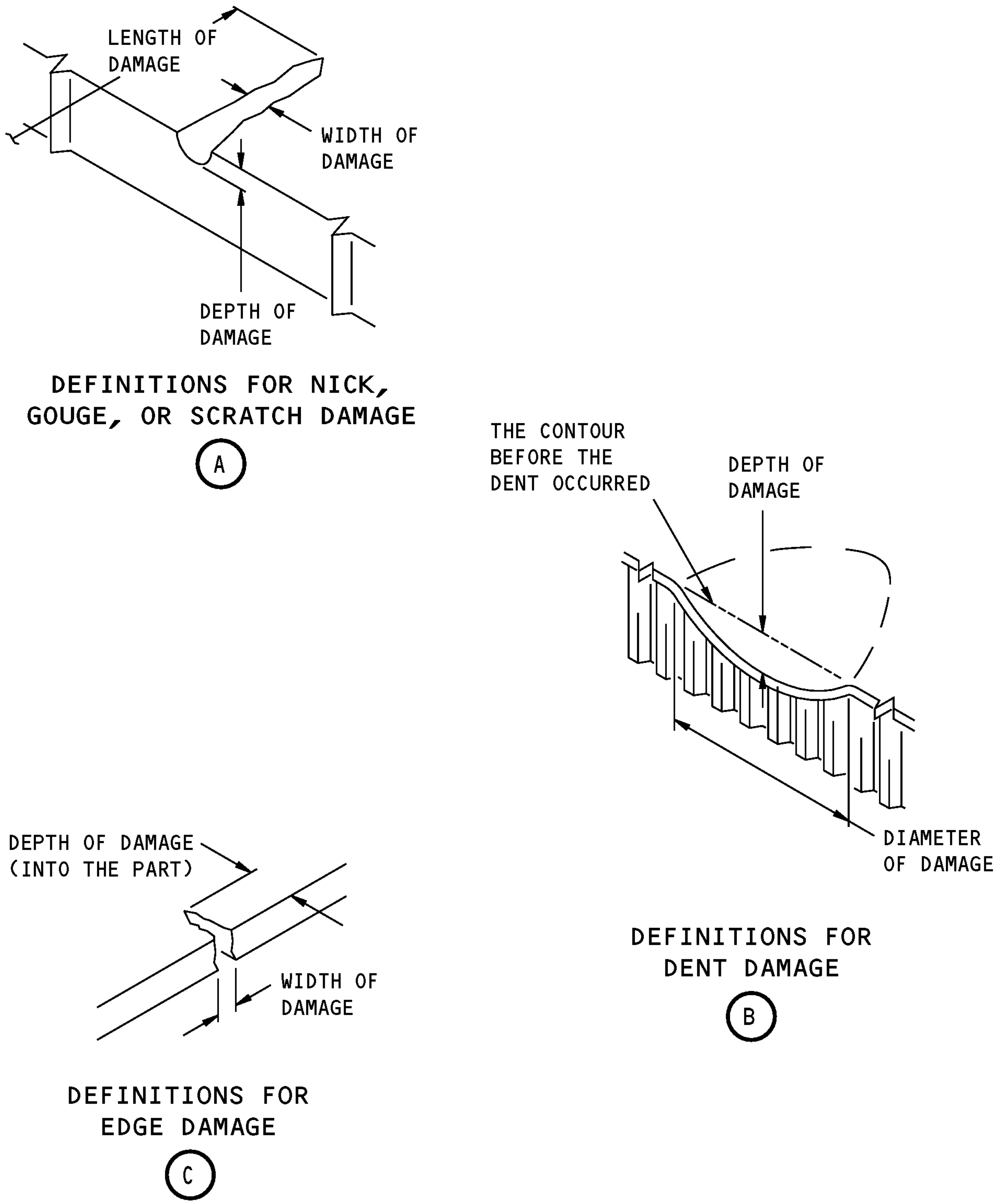
**Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage**
**Figure 103 (Sheet 2 of 3)**



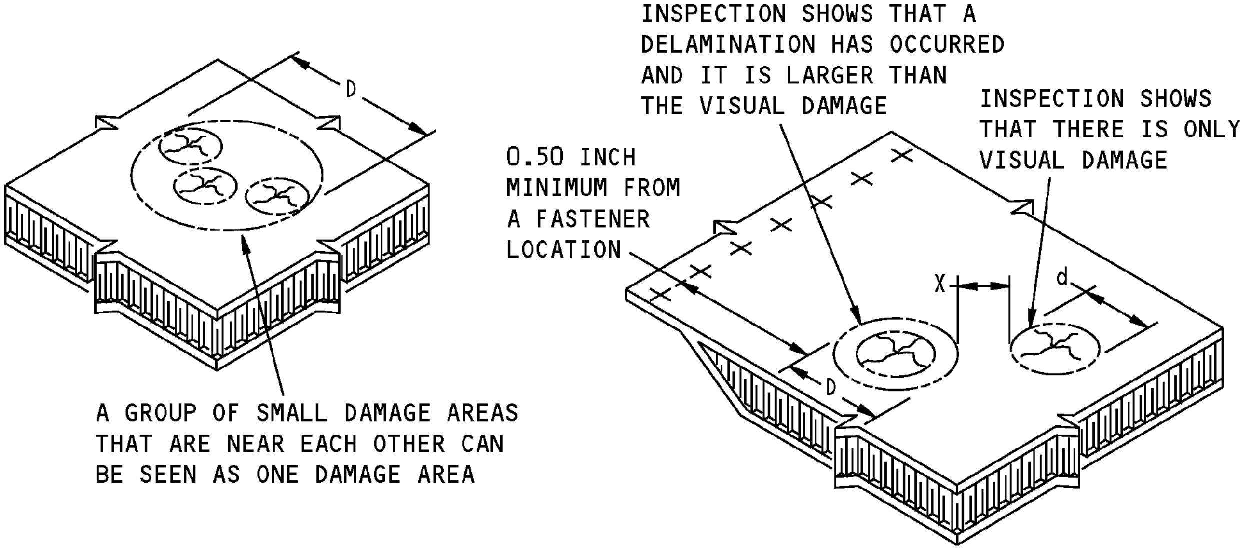

DENT THAT IS PERMITTED WITH OR WITHOUT ALUMINUM CORE

F

**Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage**
**Figure 103 (Sheet 3 of 3)**




**Damage Definitions**
**Figure 104**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH.

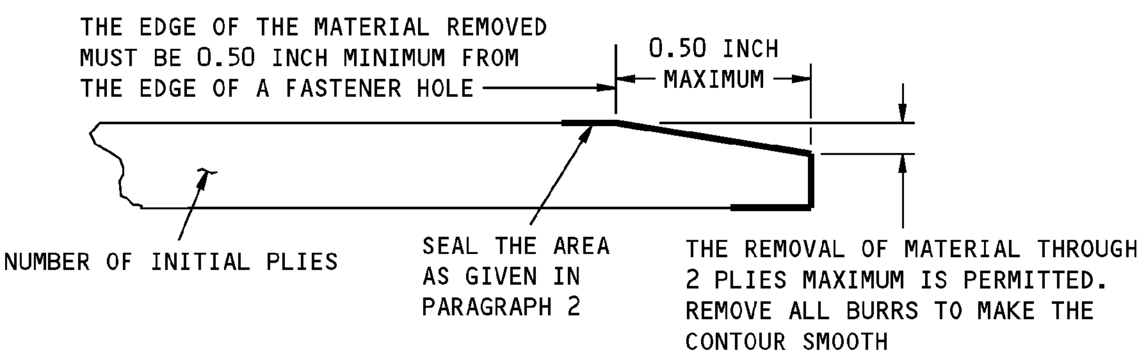
d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 2d.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A




CLEANUP AND SEALING OF EDGE EROSION

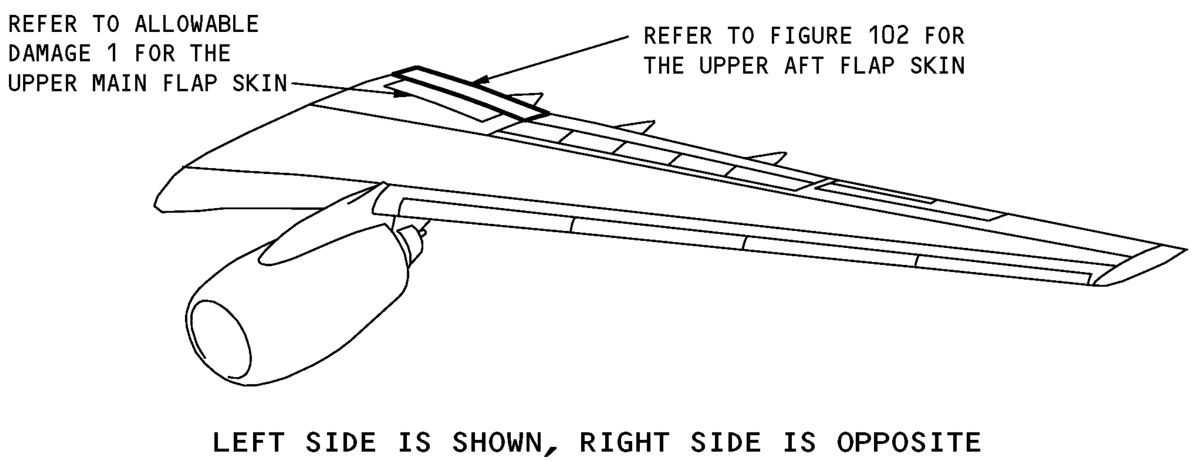
B

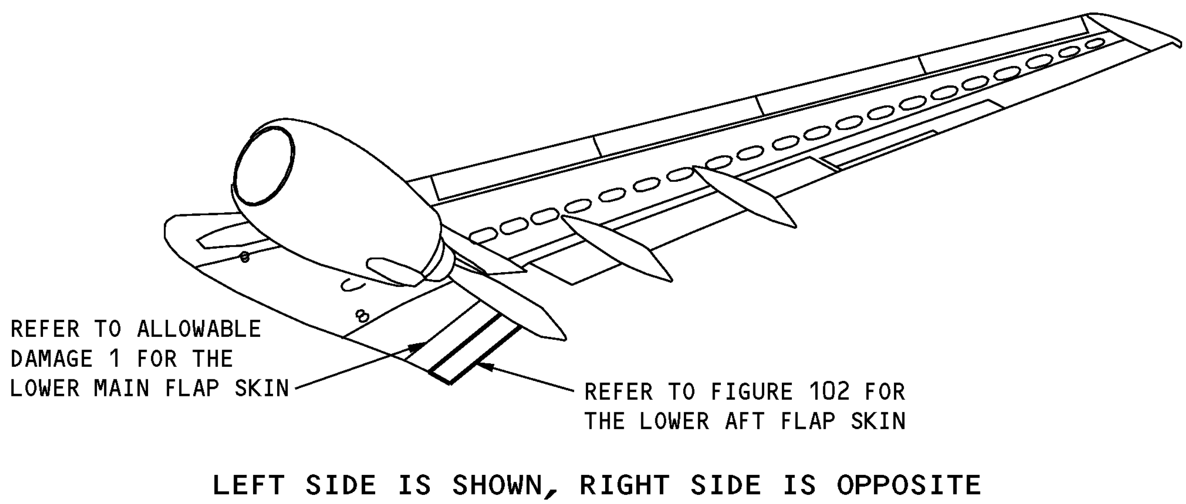
**Allowable Damage Limits for the Trailing Edge Cove (Upper Skin)**
**Figure 105**

## ALLOWABLE DAMAGE 2 - WING INBOARD TRAILING EDGE AFT FLAP

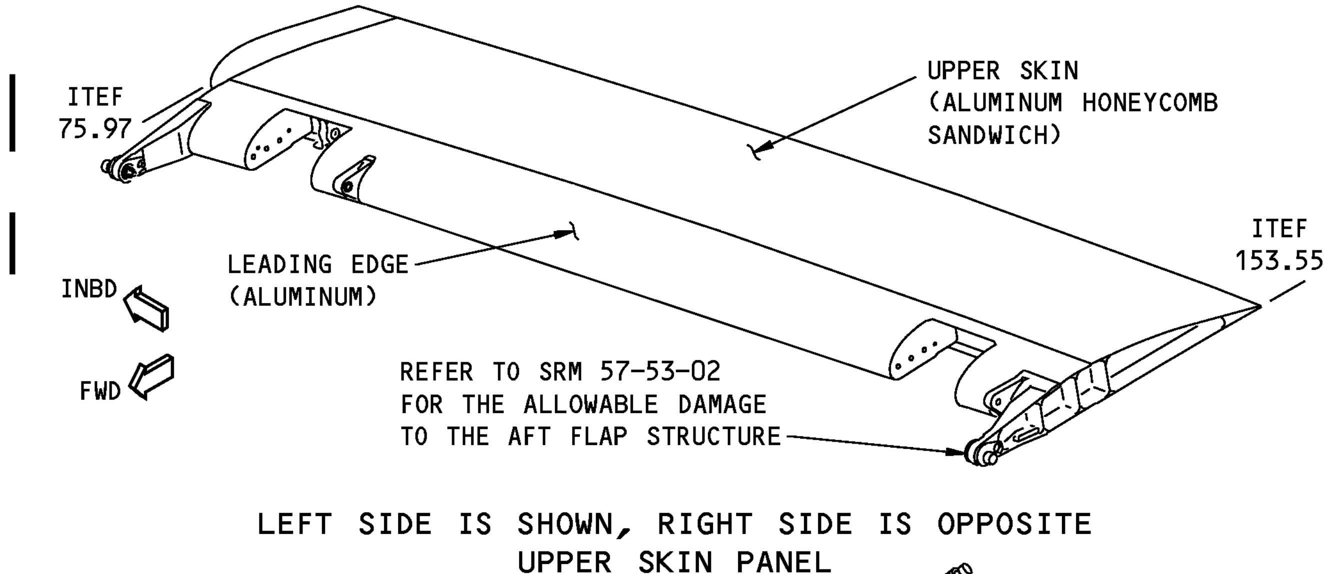
### 1. Applicability

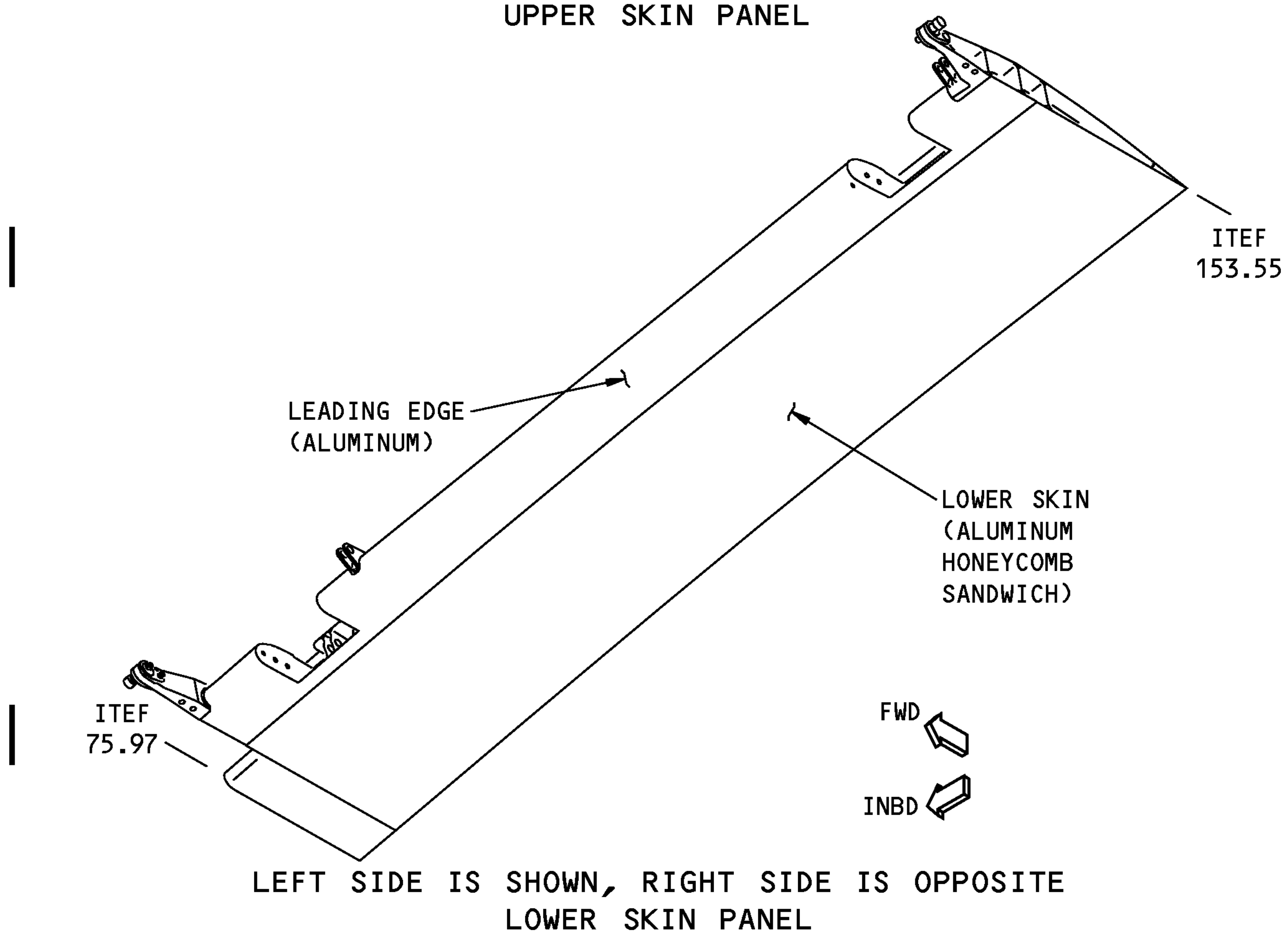
A. This subject gives the allowable damage limits for the wing inboard trailing edge aft flap shown in Wing Inboard Trailing Edge - Aft Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.




**Wing Inboard Trailing Edge - Aft Flap Skin Panel Location**
**Figure 101**




F96694 S0006594855_V2

**Wing Inboard Trailing Edge - Aft Flap Skin Panel Allowable Damage**
**Figure 102**

## 2. General

A. Do the steps that follow if you have damage:

(1) Remove the damaged material. Refer to 51-10-02 for the procedures.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you need to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

B. After you remove the damage, do as follows:

(1) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(2) Apply a layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-10 | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E.

(3) Dents:

(a) The damage is permitted as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than 0.125 inch.

b) The distance between two dents, or between a dent and other damage is less than 0.50D (D = the large dimension of the two damage areas).

2) If you fill or rework the dent, do the steps that follow:
   a) Do an inspection of the damage area each 400 flight hour interval or more frequently.
   b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures:
  (a) The damage is permitted if:
    1) It is a maximum of 0.25 inch in diameter and located in a non-critical area as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail G.
    2) You do a close visual inspection of the damage area to verify that there are no cracks or sharp edges.
    3) It is a minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.
    4) You make a temporary seal with aluminum foil tape (speed tape).

    **NOTE**: Use of vacuum and heat to remove moisture, is recommended.

      a) Keep a record of the location.
      b) Make sure the tape is in satisfactory condition and inspect at each 150 flight hour interval.
    5) You do a permanent repair before 5,000 flight hours have occurred, as specified in 51-70-10.
  (b) Holes and Punctures are not permitted in critical areas as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail G .

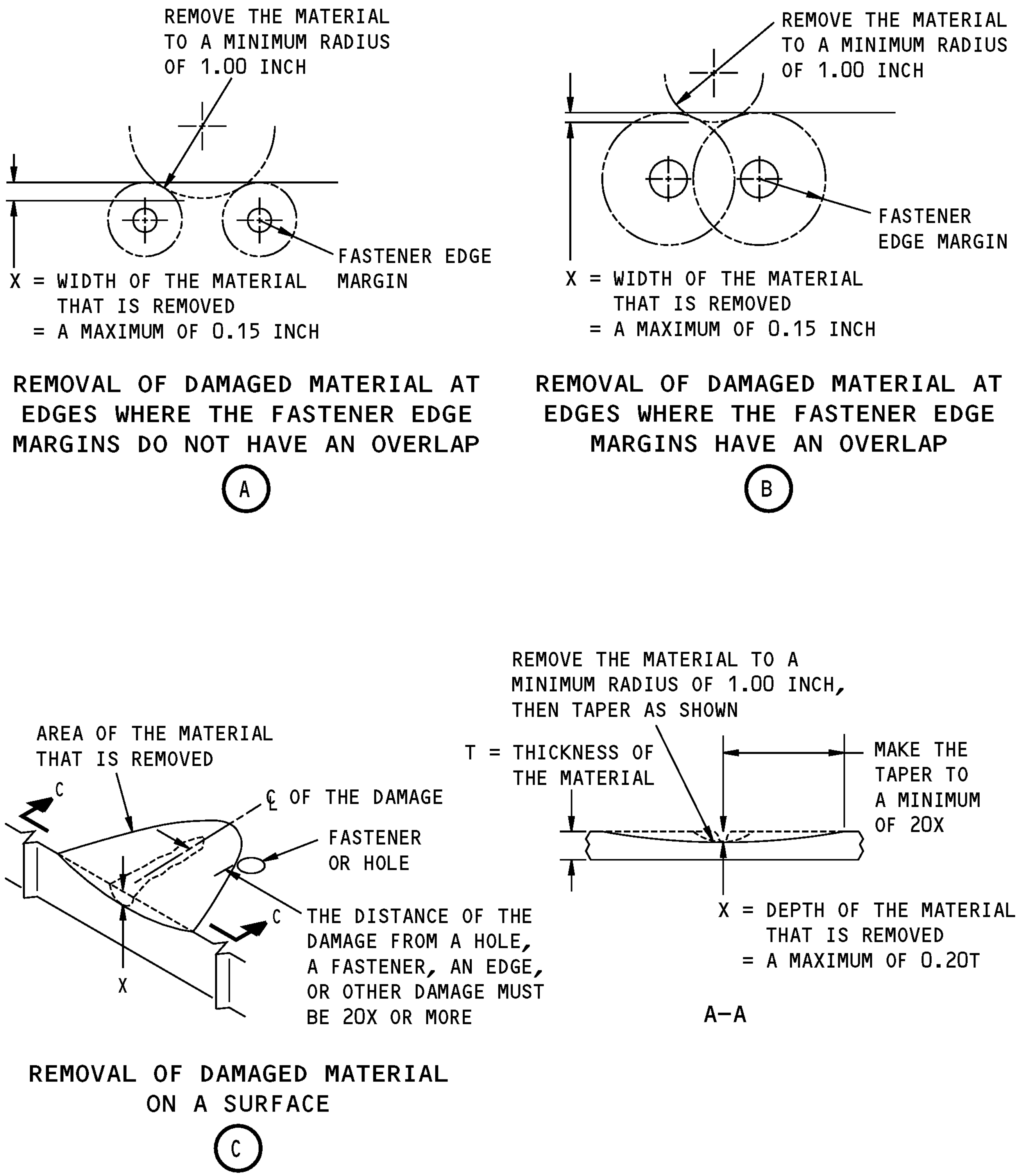
B. Leading Edge Skins (Aluminum)

(1) Cracks:
  (a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and E.

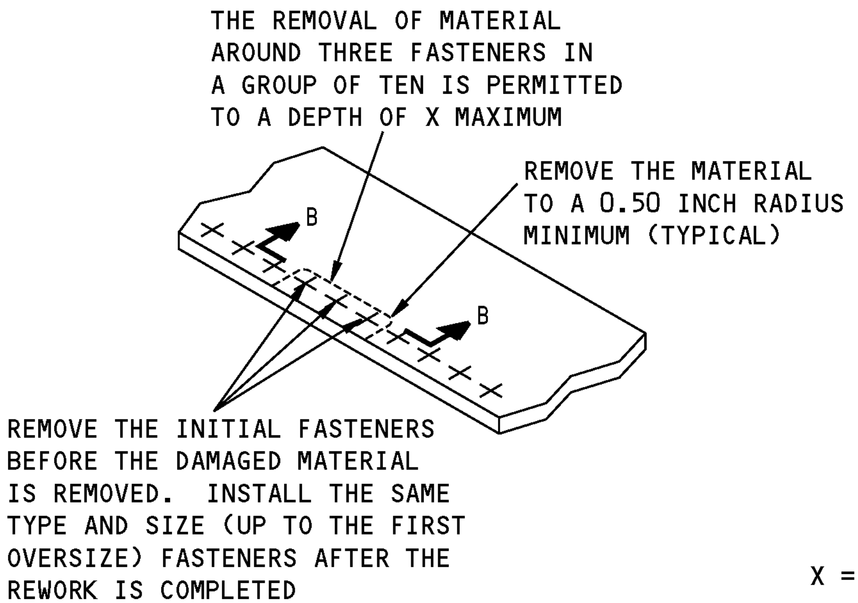
(2) Nicks, Gouges, Scratches, and Corrosion:
  (a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents:
  (a) The damage is permitted as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail F .
    1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.
      a) The depth of the dent is more than 0.125 inch.
      b) The distance between two dents, or between a dent and other damage is less than 0.50D (D = the large dimension of the two damage areas).
    2) If you fill or rework the dent, do the steps that follow:
      a) Do an inspection of the damage area each 400 flight hour interval or more frequently.
      b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures are not permitted.

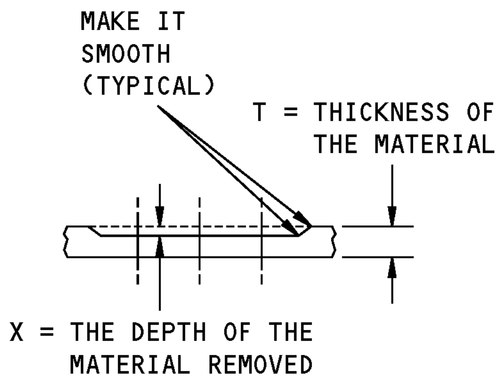



**Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage**
**Figure 103 (Sheet 1 of 4)**

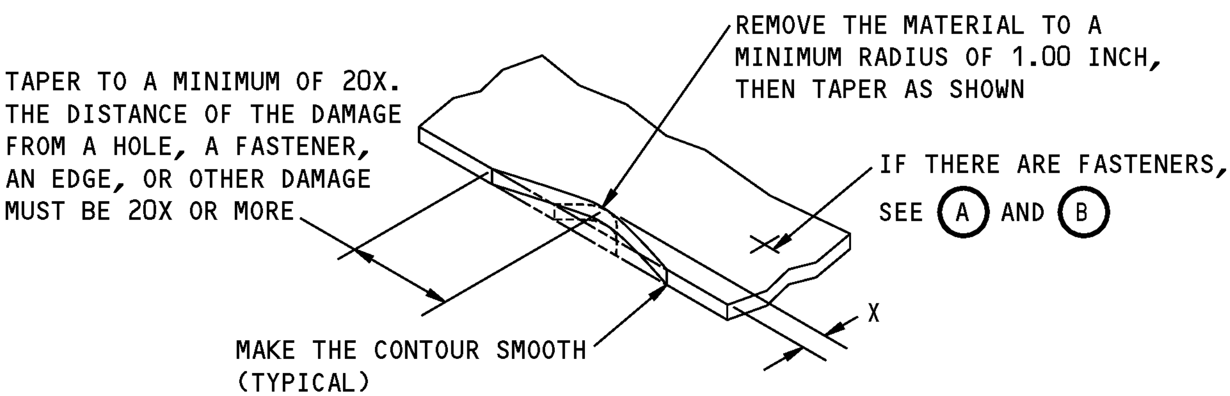



REMOVAL OF CORROSION AROUND THE FASTENERS

D




B-B

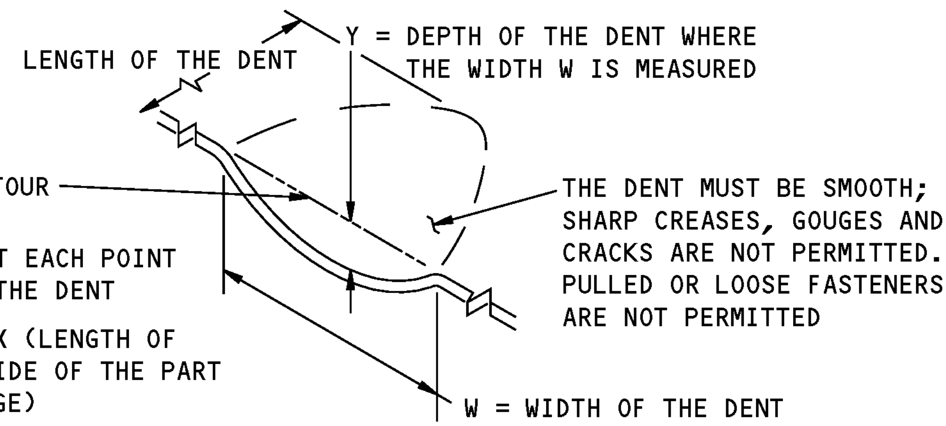



X = WIDTH OF THE MATERIAL THAT IS REMOVED = A MAXIMUM OF 0.15 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

E

**Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage**
**Figure 103 (Sheet 2 of 4)**




$\frac{W}{Y}$ MUST BE 10 OR MORE AT EACH POINT ALONG THE LENGTH OF THE DENT

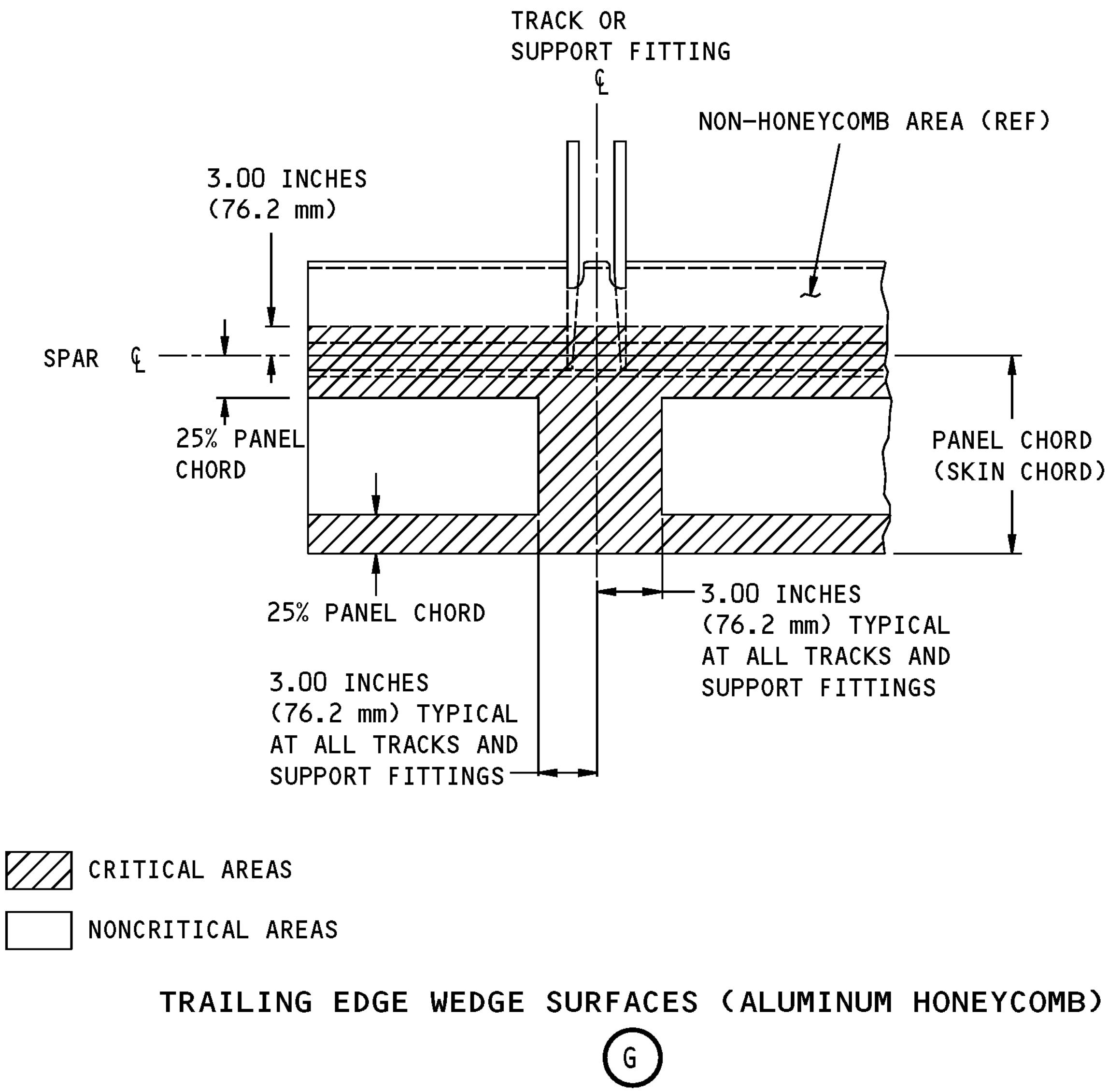
W = A MAXIMUM OF 0.25 X (LENGTH OF THE PANEL ON THE SIDE OF THE PART WHICH HAS THE DAMAGE)

Y = A MAXIMUM OF 0.25 INCH OR 0.25 X (THE DEPTH OF THE HONEYCOMB CORE) THAT WHICH IS LESS

DENT THAT IS PERMITTED

F

**Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage**
**Figure 103 (Sheet 3 of 4)**

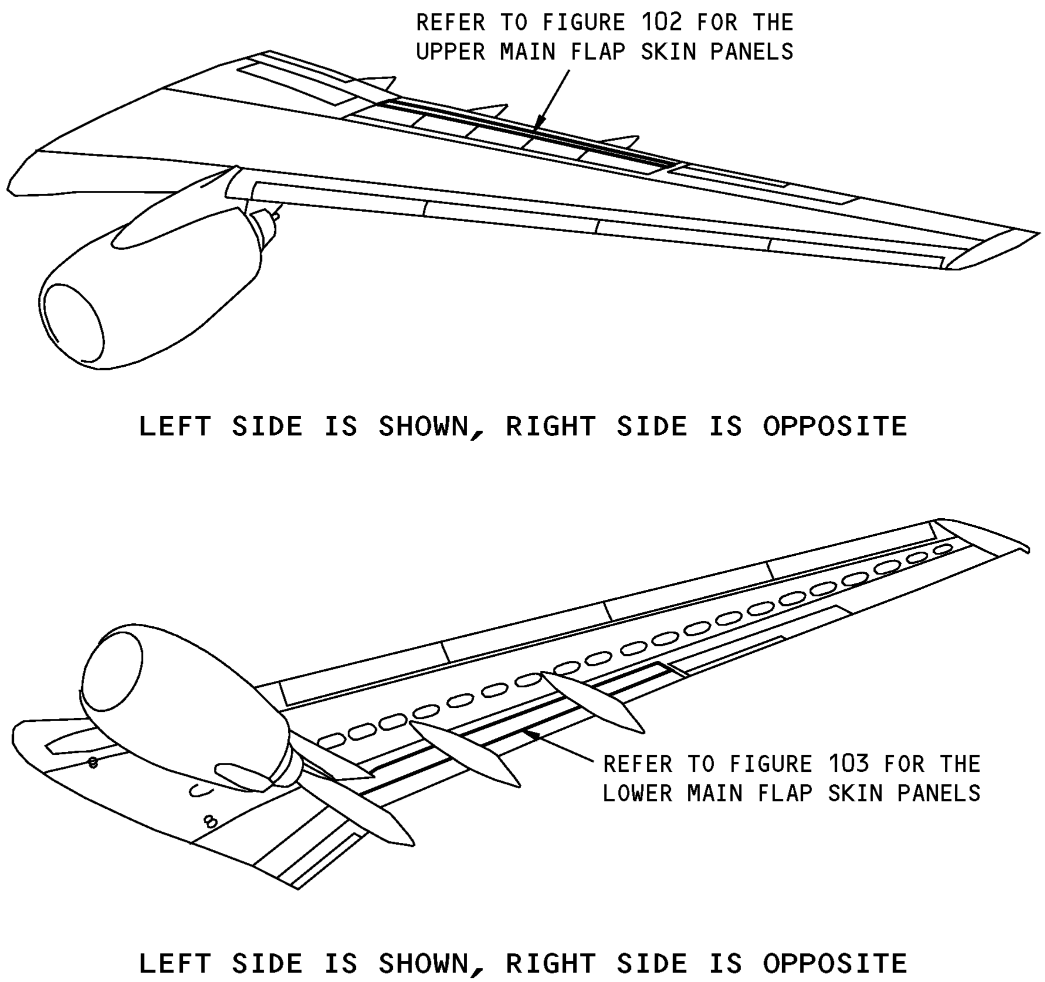



**Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage**
**Figure 103 (Sheet 4 of 4)**

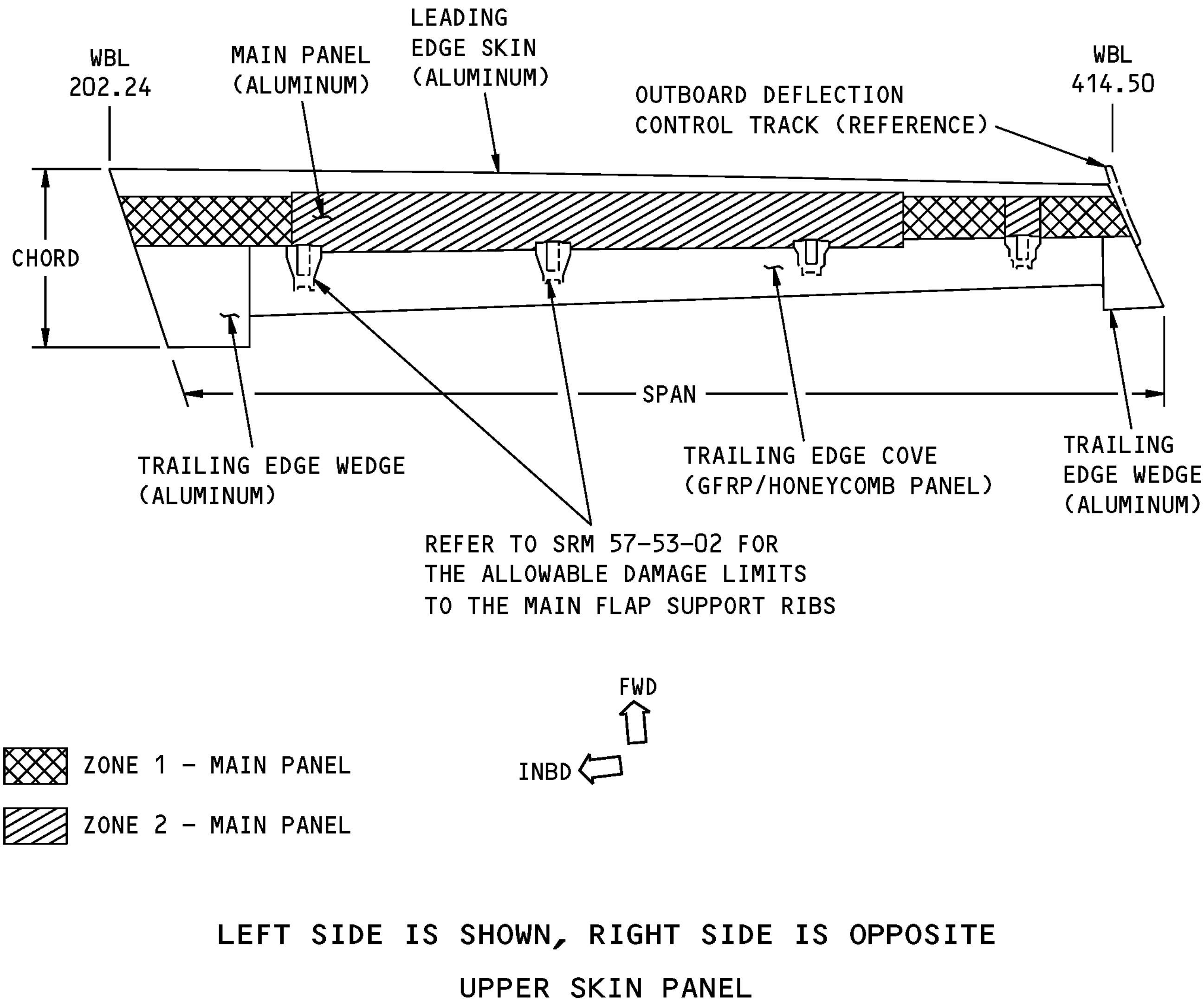
## ALLOWABLE DAMAGE 3 - WING OUTBOARD TRAILING EDGE MAIN FLAP SKIN

1. **Applicability**

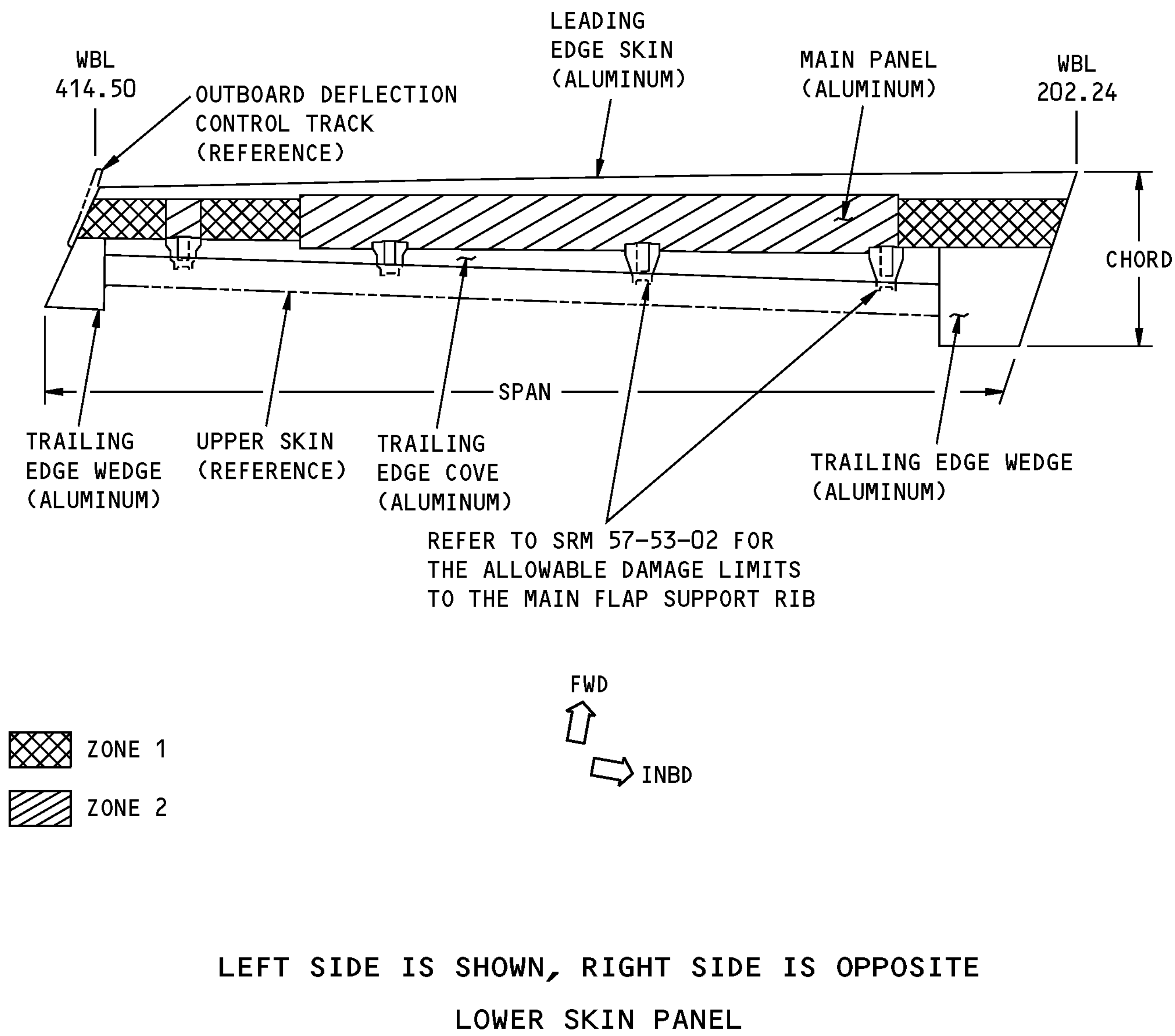
   A. This subject gives the allowable damage limits for the wing outboard trailing edge main flap skin shown in Wing Trailing Edge Main Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 3.




**Wing Trailing Edge Main Flap Skin Panel Location**
**Figure 101**




**Wing Outboard Trailing Edge Upper Main Flap Skin Zones**
**Figure 102**




**Wing Outboard Trailing Edge Lower Main Flap Skin Zones**
**Figure 103**

## 2. General

A. Do the steps that follow if you have damage to the aluminum skins:

(1) If you have to remove the damage, refer to 51-10-02 for the procedures.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do as follows:

(1) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(2) Apply one layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

C. Do the steps that follow if you remove damage from the trailing edge cove made of Glass Fiber Reinforced Plastic (GFRP) on the upper skin panel.

(1) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits in Paragraph 4./ALLOWABLE DAMAGE 3.

(a) Make a temporary seal.

1) Apply aluminum foil tape (speed tape).

2) Keep a record of the location.

3) Make sure the tape is in satisfactory condition at each 400 flight hour interval or more frequently.

4) Seal the damage permanently no later than 5000 flight hours.

(b) Make a permanent seal.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to AMM 51-21-00/701.

3) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(2) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 3.

(a) Use vacuum and heat to remove moisture from the solid laminate and the honeycomb core. Refer to 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Repair the damage no later than 400 flight hours.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| TYPE OF PANEL | ZONE LOCATION | PARAGRAPH |
| Main Panels | 1 | 4.A |
| | 2 | 4.B |

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| TYPE OF PANEL | ZONE LOCATION | PARAGRAPH |
| Leading Edge Skin and Trailing Edge Cove Lower Skin | – | 4.C |
| Trailing Edge Wedge | – | 4.D |
| Trailing Edge Cove Upper Skin | – | 4.E |

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| 51-70-10 | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| AMM 51-21-00/701 | Interior And Exterior Finishes - Cleaning/Painting |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Main Panels (Upper and Lower Skins) - Zone 1

(1) Cracks:

(a) Remove the damage as shown in Wing Outboard Main Flap - Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 3, Details A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Outboard Main Flap - Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:

(a) The damage is permitted as shown in Wing Outboard Main Flap - Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 3, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch.

b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:
   a) Do an inspection of the damage area each 400 flight hour interval or more frequently.
   b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

B. Main Panels (Upper and Lower Skins) - Zone 2

(1) Cracks:
   (a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details, A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:
   (a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:
   (a) The damage is permitted as shown in Figure 104/ALLOWABLE DAMAGE 3, Detail F.
      1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.
         a) The depth of the damage is more than 0.10 inch.
         b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).
      2) If you fill or rework the damage, do the steps that follow:
         a) Do an inspection of the damage area each 400 flight hour interval or more frequently.
         b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

C. Leading Edge Skin and Trailing Edge Cove (Lower Skin)

(1) Cracks:
   (a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details, A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:
   (a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:
   (a) The damage is permitted as shown in Figure 104/ALLOWABLE DAMAGE 3, Detail F.
      1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.
         a) The depth of the damage is more than 0.10 inch.
         b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).
      2) If you fill or rework the damage, do the steps that follow:
         a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

D. Trailing Edge Wedge (Upper and Lower Skins)

(1) Cracks:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details, A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:

(a) The damage is permitted as shown in Figure 104/ALLOWABLE DAMAGE 3, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than 0.10 inch

b) The distance between two dents or between a dent and other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures are not permitted.

E. Trailing Edge Cove (Upper Skin)

**NOTE**: For edge damage only, refer to Damage Definitions, Figure 105/ALLOWABLE DAMAGE 3, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Nicks, gouges, and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges, and scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply.

(b) A maximum of 3.0 inches in length

(c) A maximum of 0.25 inch in width

(d) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

(3) Dents are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

**NOTE**: Use the limits for holes and punctures if there is glass fiber damage.

(b) The damage is a minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch in diameter

(b) A maximum of one facesheet and the core in depth

(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

(5) Delaminations are permitted as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A if they are:

(a) A maximum of 0.50 inch in length

(b) A maximum of 0.50 inch in width.

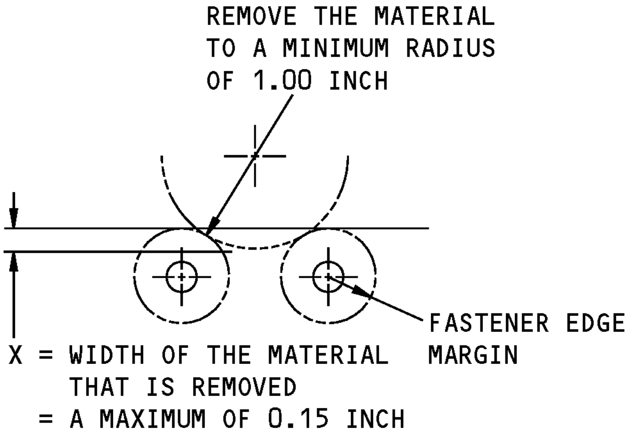
(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

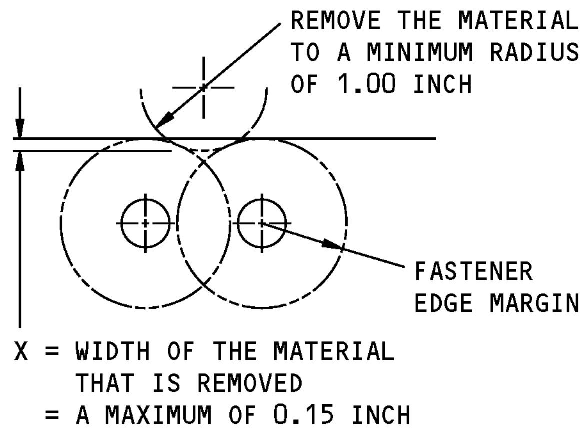
(b) A maximum of 0.25 inch in width.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail B.



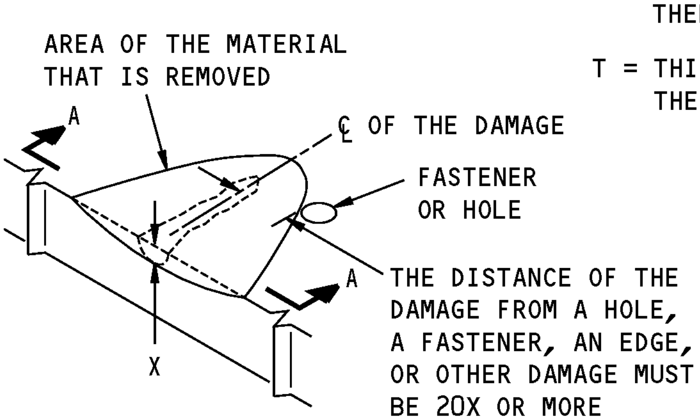

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A



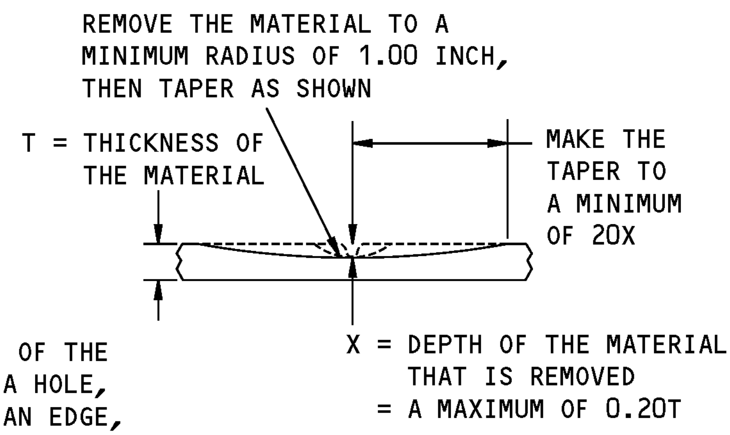

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B



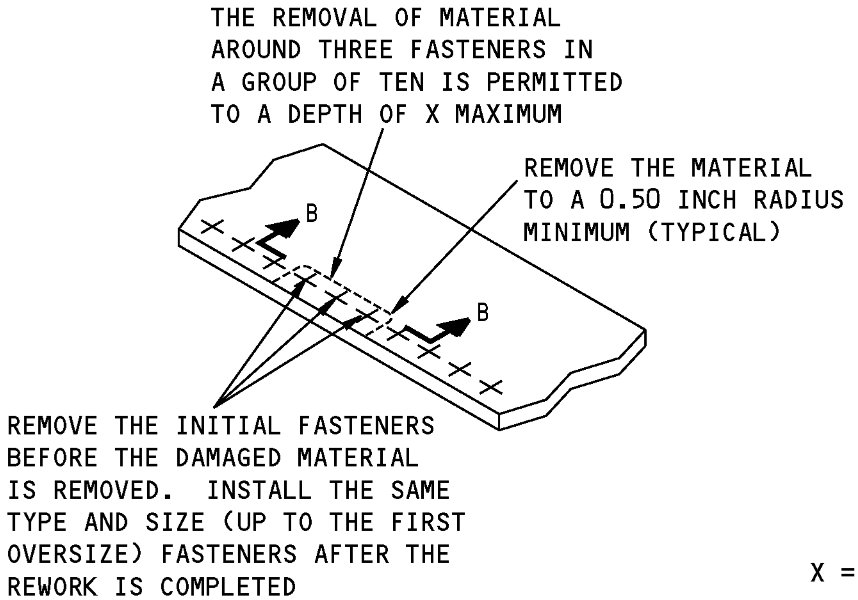

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C



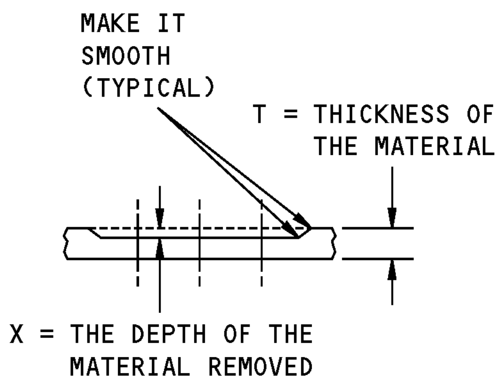

A-A

**Wing Outboard Main Flap - Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**

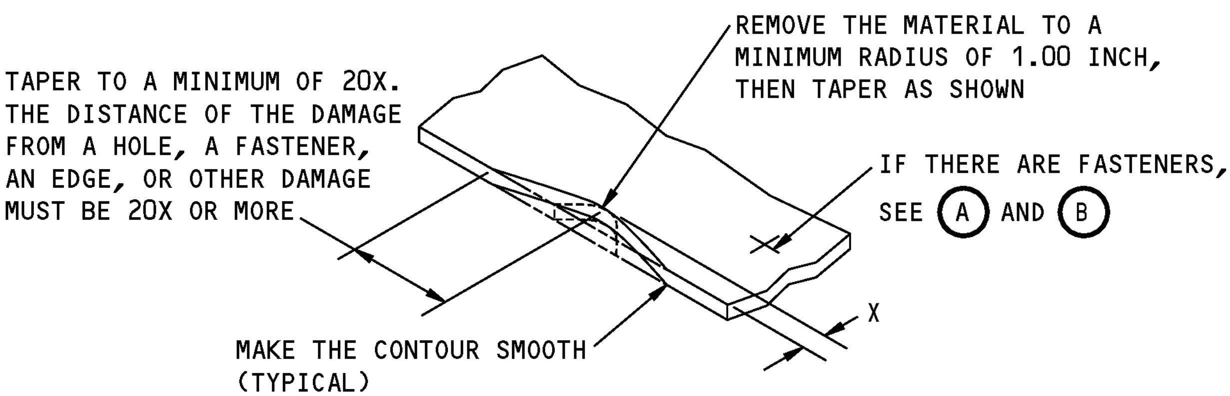



REMOVAL OF CORROSION
AROUND THE FASTENERS

D




B-B

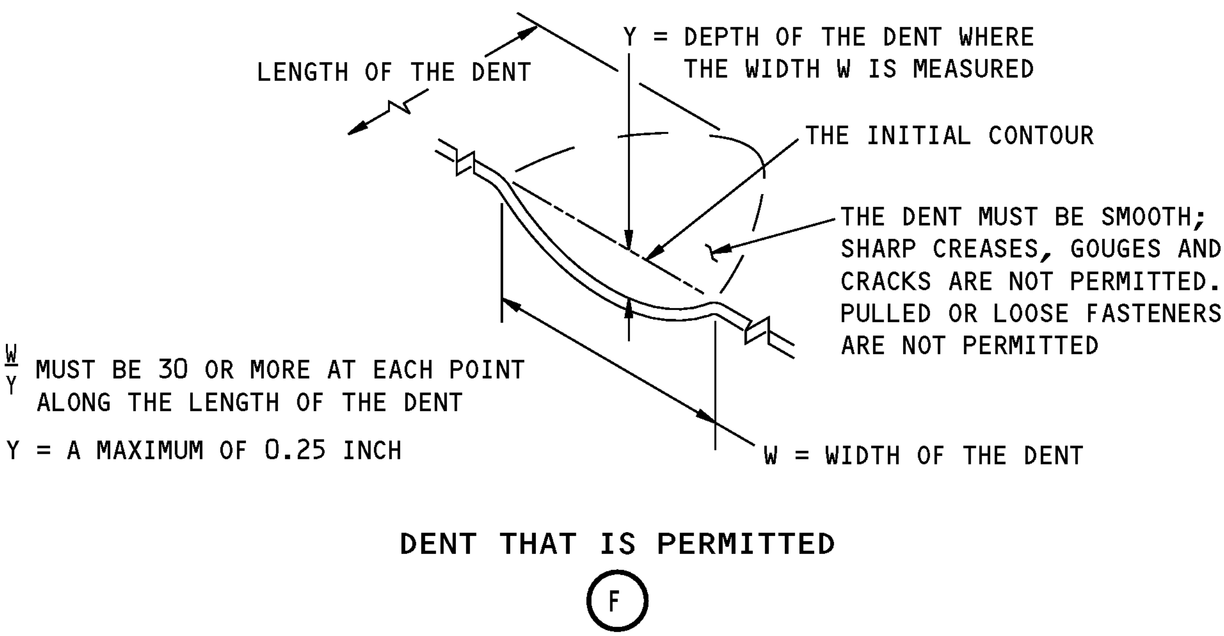



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.15 INCH

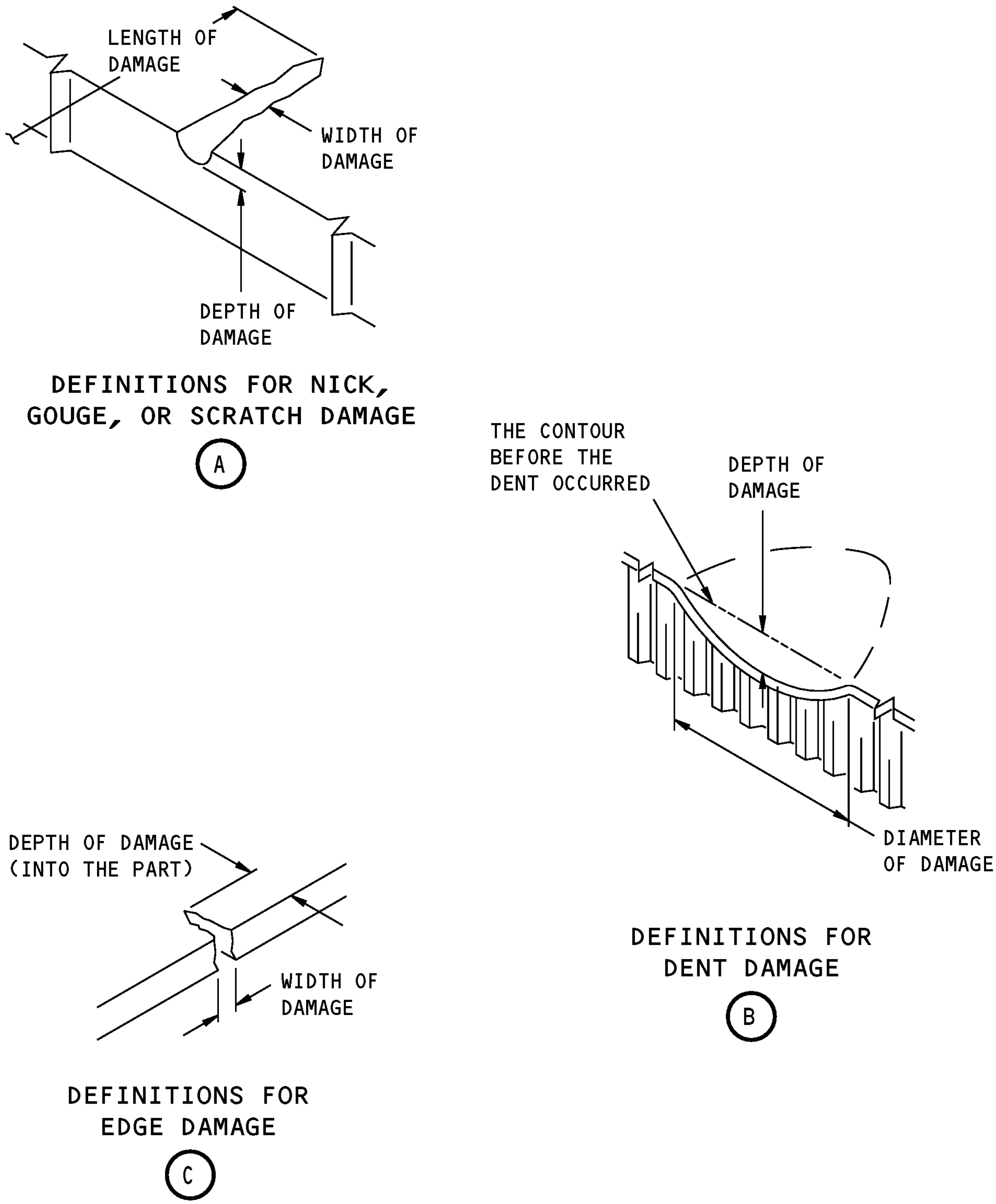
REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

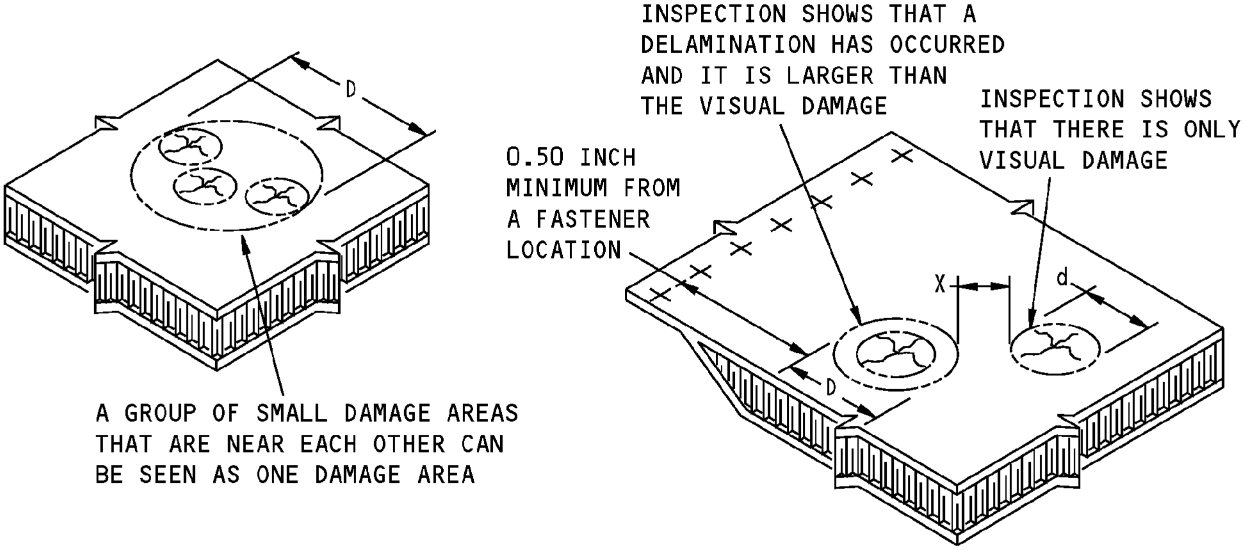
**Wing Outboard Main Flap - Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**




**Wing Outboard Main Flap - Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**




**Damage Definitions**
**Figure 105**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH.

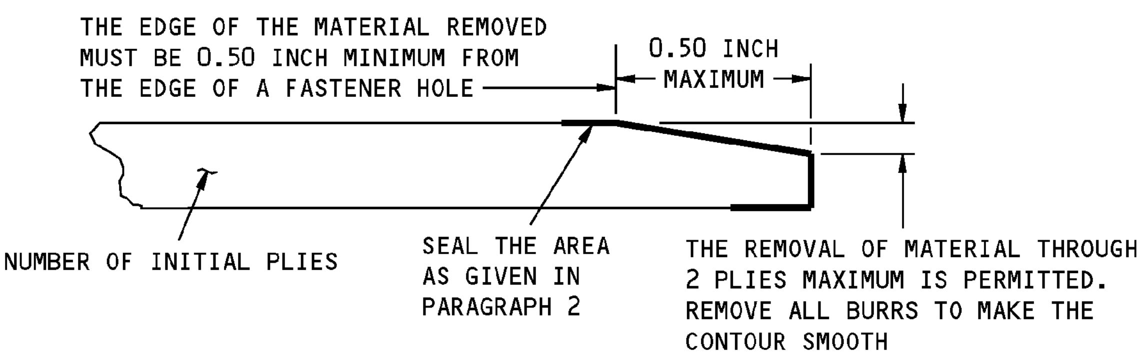
d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 2d.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A




CLEANUP AND SEALING OF EDGE EROSION

B

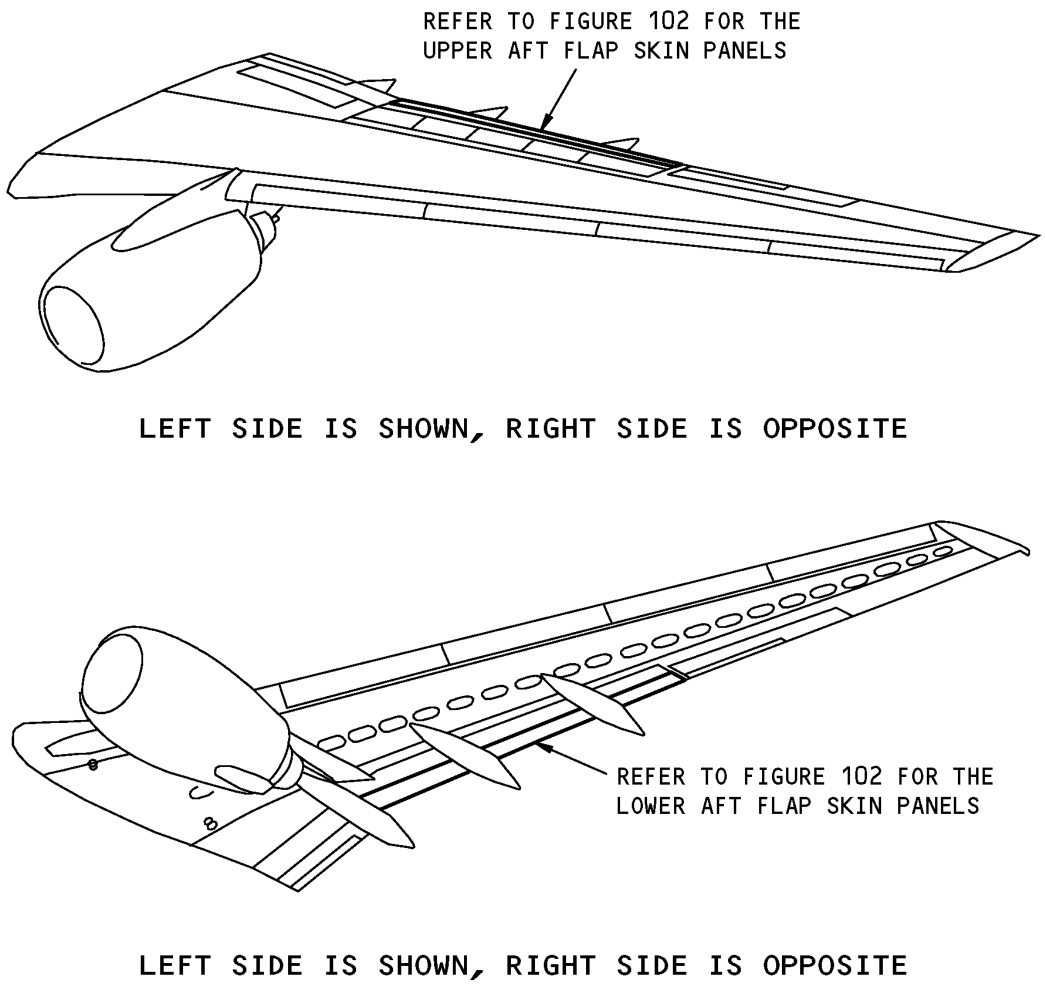
**Allowable Damage Limits for the Trailing Edge Cove (Upper Skin)**
**Figure 106**

## ALLOWABLE DAMAGE 4 - WING OUTBOARD TRAILING EDGE AFT FLAP

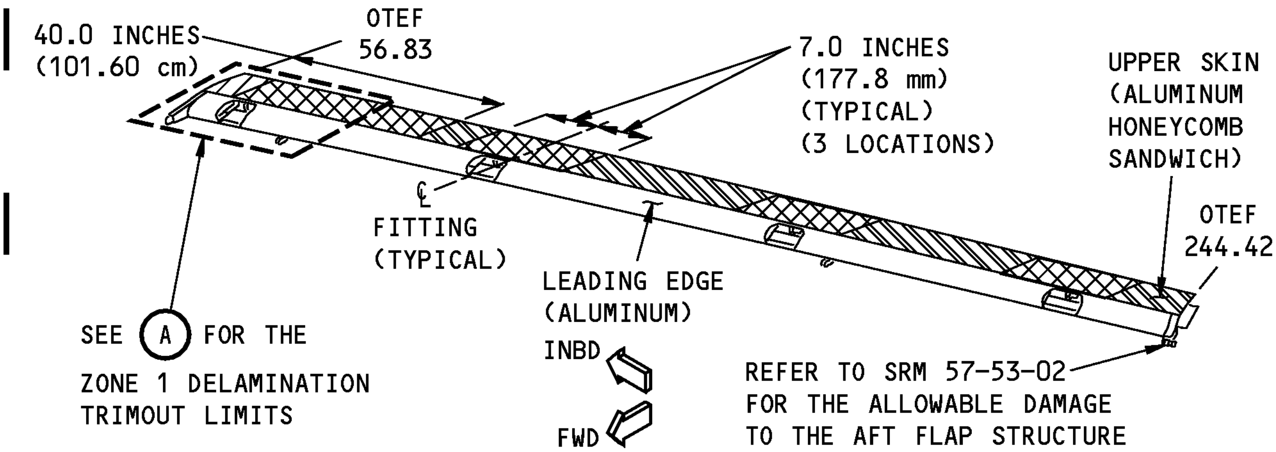
### 1. Applicability

A. This subject gives the allowable damage limits for the wing outboard trailing edge aft flap as shown in Wing Trailing Edge Aft Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 4.

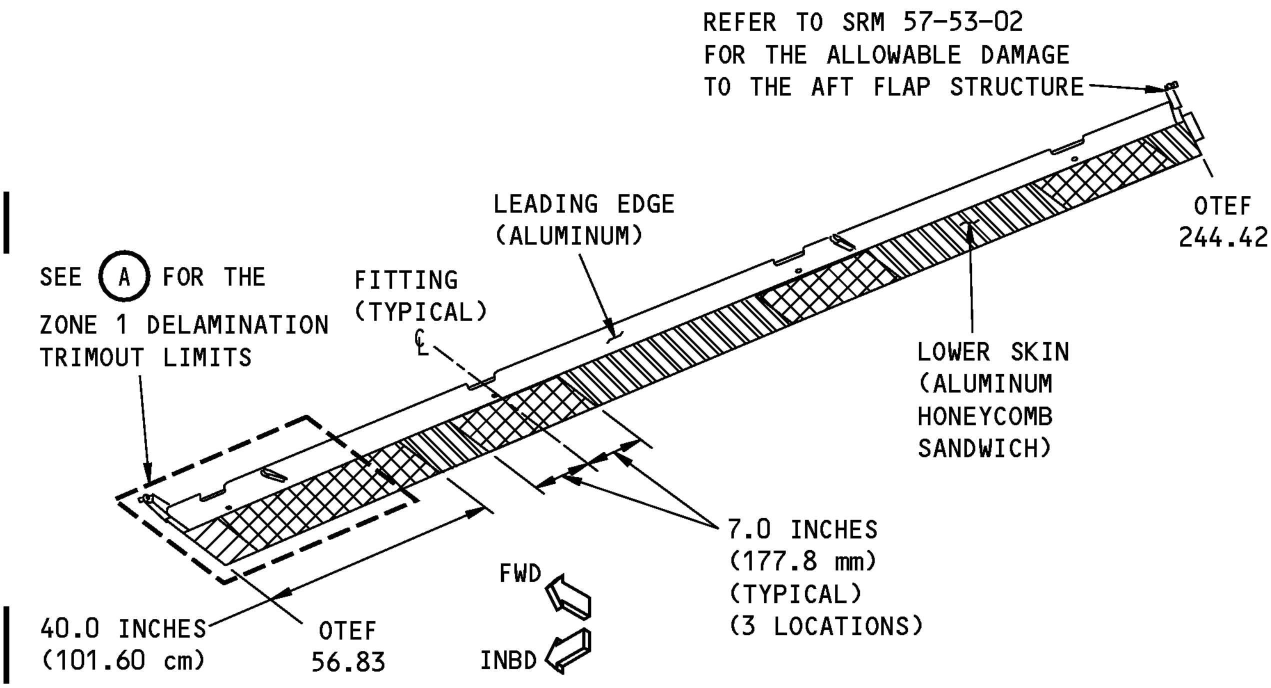
B. Allowable Damage 4 is only applicable to wing outboard trailing edge aft flaps which have initial 250 degree F (121 degree C) cure trailing edge wedge assemblies. Refer to drawing 113A3700-1 thru -22 for the 250 degree F (121 degree C) cure trailing edge aft flap assembly details.




**Wing Trailing Edge Aft Flap Skin Panel Location**
**Figure 101**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL

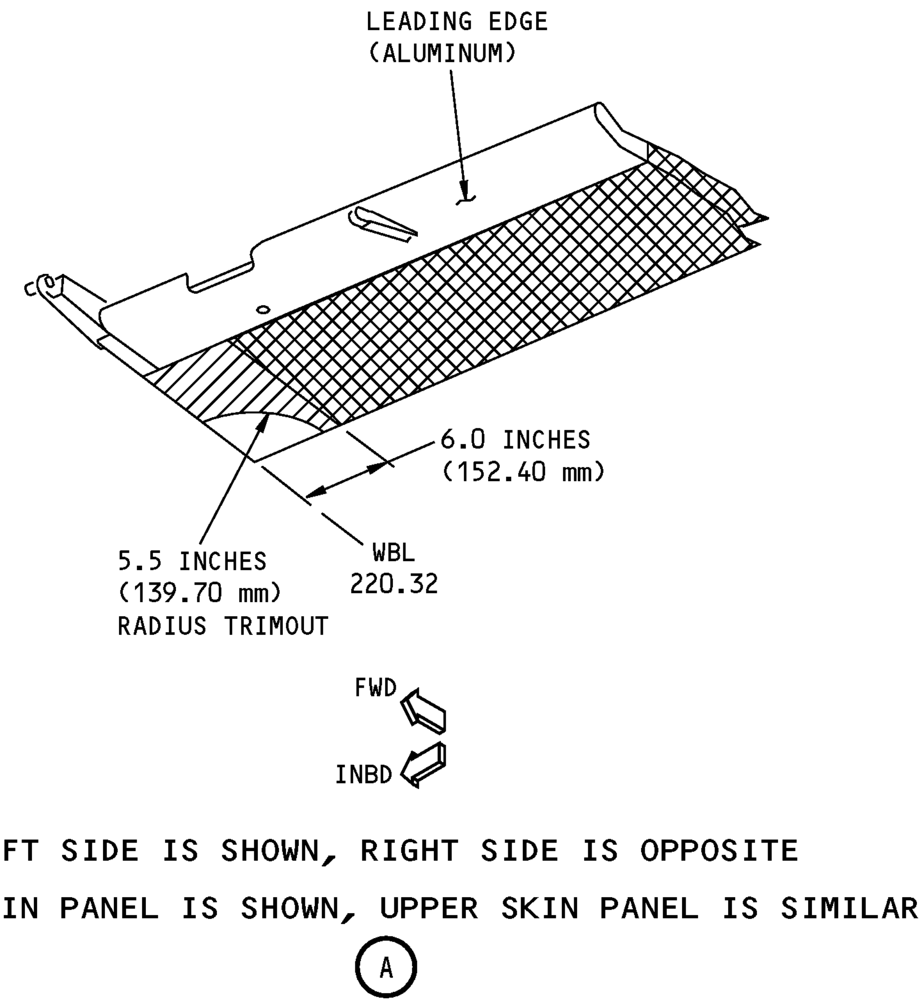



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

ZONE 1 - DELAMINATION LIMITS
ZONE 2 - DELAMINATION LIMITS
ZONE 3 - DELAMINATION LIMITS

F96839 S0006594881_V3

**Wing Outboard Trailing Edge - Aft Flap Skin Panel Location**
**Figure 102 (Sheet 1 of 2)**




**Wing Outboard Trailing Edge - Aft Flap Skin Panel Location**
**Figure 102 (Sheet 2 of 2)**

2. General
   A. Refer to Paragraph 4./ALLOWABLE DAMAGE 4 for the allowable damage limits.
   B. Remove the damage as necessary.
      (1) Refer to 51-10-02 for the investigation and removal of damage.
      (2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.
      (3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.
   C. After you remove the damage, do the steps that follow:
      (1) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.
      (2) Apply a layer of BMS 10-79, Type III primer to the reworked areas. Refer to SOPM 20-44-04.
      (3) Apply a layer of BMS 10-60, Type II enamel to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.
   D. Do the steps that follow if you have delamination to the upper or lower skin in the Zone 1 location as shown in Figure 102 for the trimout location.
      (1) Trimout the skin, core, end rib, seal, and fiberglass aft filler to a maximum 5.5 inches (139.70 mm) radius. Refer to Figure 102 for the trimout location. Make sure there is a minimum 1.5D (where D = fastener diameter) edge margin to any adjacent fasteners.
      (2) Extend the trimout of the core 0.20 inch (5.08 mm) beyond the trimout to the skin (trim the fiberglass aft filler and the inboard end rib flush with the skin trimout). Fill the trimout of the core flush to the skin trimout at all locations. Fill all gaps and apply a fillet seal to all skin-to-core bond locations with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05, GENERAL.

3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05, GENERAL | Repair Sealing |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-10 | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. Allowable Damage Limits
   A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)
      (1) Cracks:
         (a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details, A, B, and E.
      (2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail F .

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

b) The distance away from other damage is less than 0.50D ( D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted if:

1) It is a maximum of 0.25 inch in diameter and located in a non-critical area as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail G .

2) You do a close visual inspection of the damage to verify that there are no cracks or sharp edges.

3) It is a minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

4) You make a temporary seal with aluminum foil tape (speed tape).

**NOTE**: Use of vacuum and heat to remove moisture, is recommended.

a) Keep a record of the location.

b) Make sure the tape is in satisfactory condition and inspect at each 150 flight hour interval.

5) You do a permanent repair before 5,000 flight hours have occurred, as specified in 51-70-10.

(b) Holes and Punctures are not permitted in critical areas as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail G .

(5) Zone 1 - Upper and Lower Skin Delamination

(a) If you do not trim out the damage as given in Paragraph 2 and Figure 102 you may apply a temporary seal if you do one of the two steps that follow:

1) If the damage is located as shown in Figure 102, Sheet 2 and not more than 5.50 inches (139.70 mm) in radius from the inboard aft edge then seal the damage with aluminum foil tape (speed tape).

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

2) If the damage is not located as shown in Figure 102, Sheet 2 but is located in Zone 1 and is not more than 2.00 inches (50.80 mm) in diameter and greater than 1.00 inch (25.4 mm) away from an edge, then seal the damage with aluminum foil tape (speed tape). Make sure that all adjacent damage in Zone 1 is as shown in Figure 103, Detail H.

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(6) Zone 2 - Upper and Lower Skin Delamination

(a) The damage is permitted as given in Figure 103, Detail H if you do the steps that follow:

1) If the damage is not more than 1.50 inches (38.10 mm) in diameter and greater than 1.00 inch (25.4 mm) away from an edge then seal the damage with aluminum foil tape (speed tape).

2) Keep a record of the damage location.

3) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

4) Replace the tape as necessary.

5) Repair the damage if the damage becomes larger.

6) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(7) Zone 3 - Upper and Lower Skin Delamination

(a) The damage is permitted as given in Figure 103, Detail H if you do the steps that follow:

1) If the damage is not more than 0.50 inch (12.70 mm) in diameter and greater than 1.00 inch (25.4 mm) away from an edge then seal the damage with aluminum foil tape (speed tape).

2) Keep a record of the damage location.

3) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

4) Replace the tape as necessary.

5) Repair the damage if the damage becomes larger.

6) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

B. Leading Edge Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail F .

1) If the damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

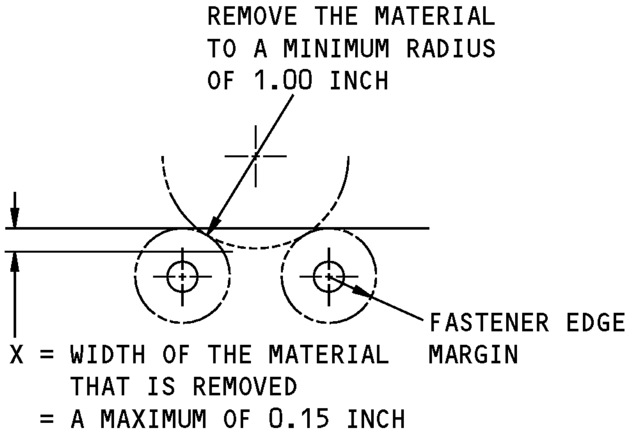
b) The distance away from other damage is less than 0.50D ( D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

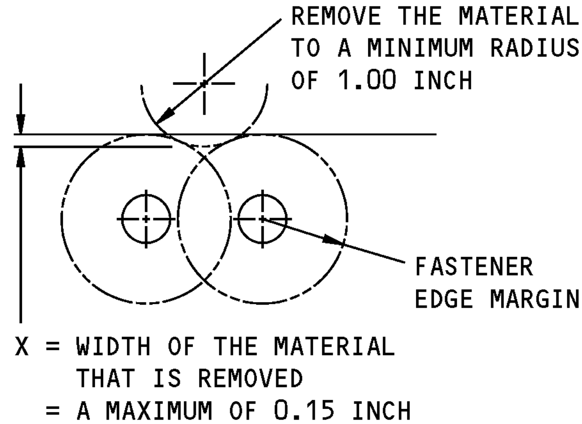
b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.



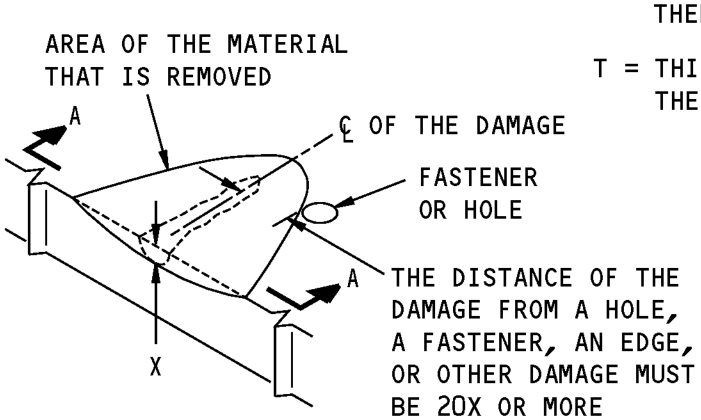

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

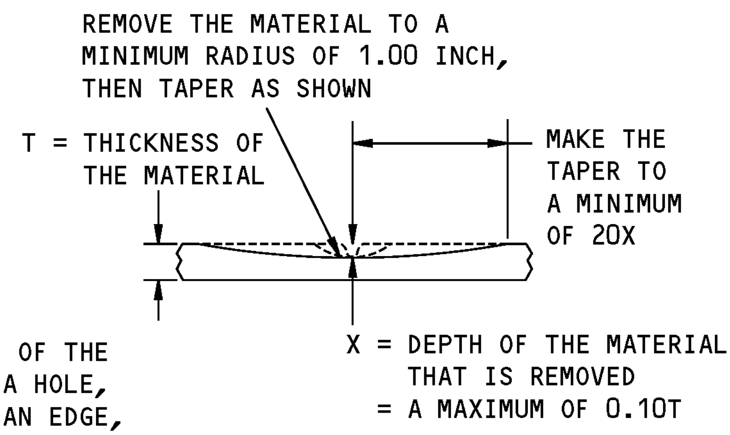
A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B



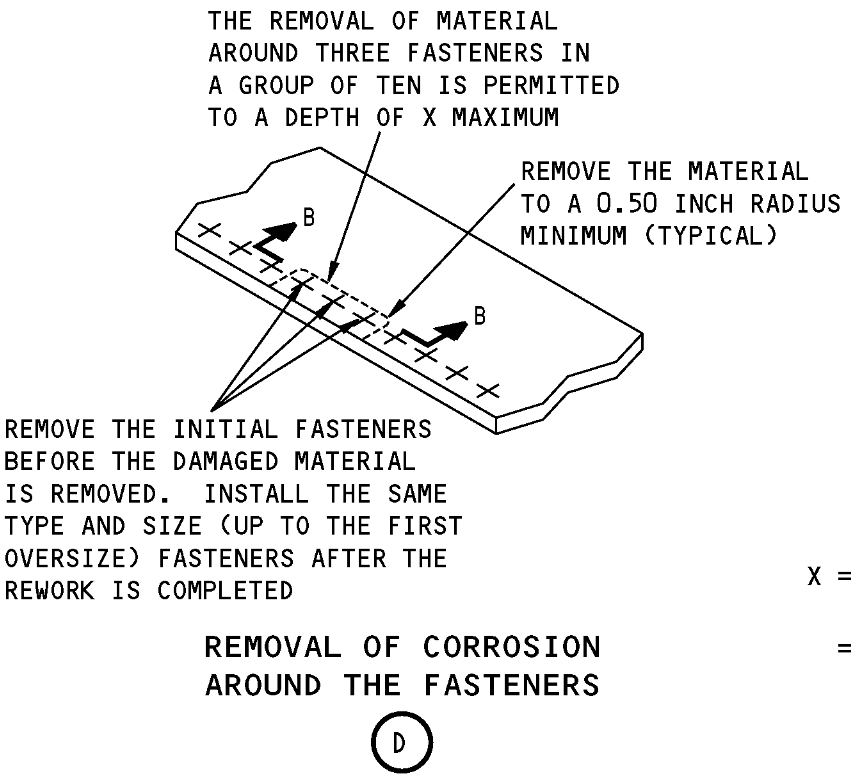



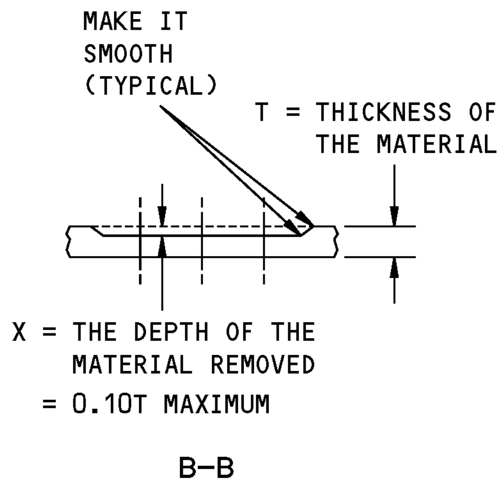

A-A

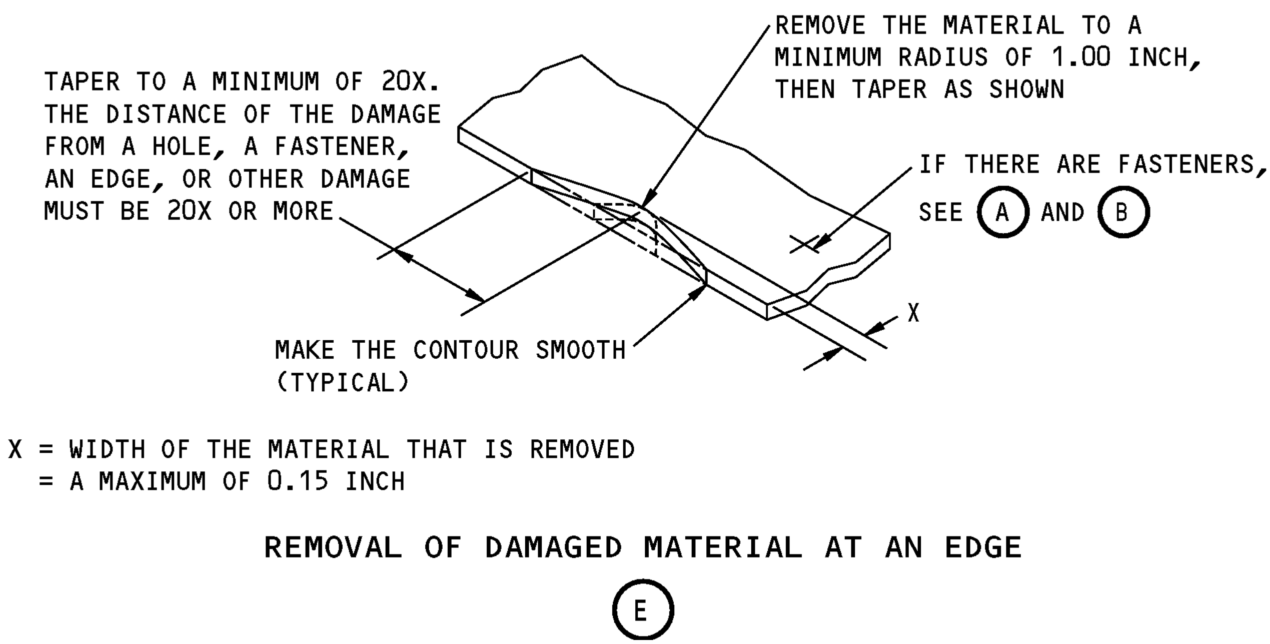
REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

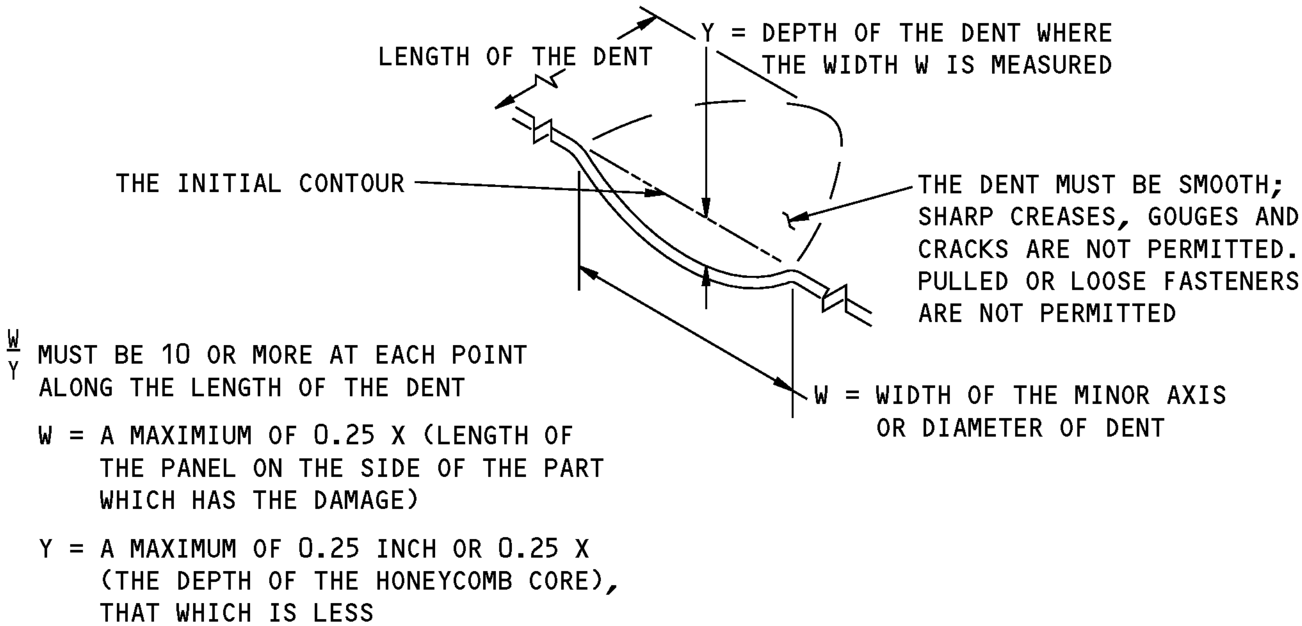
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 5)**



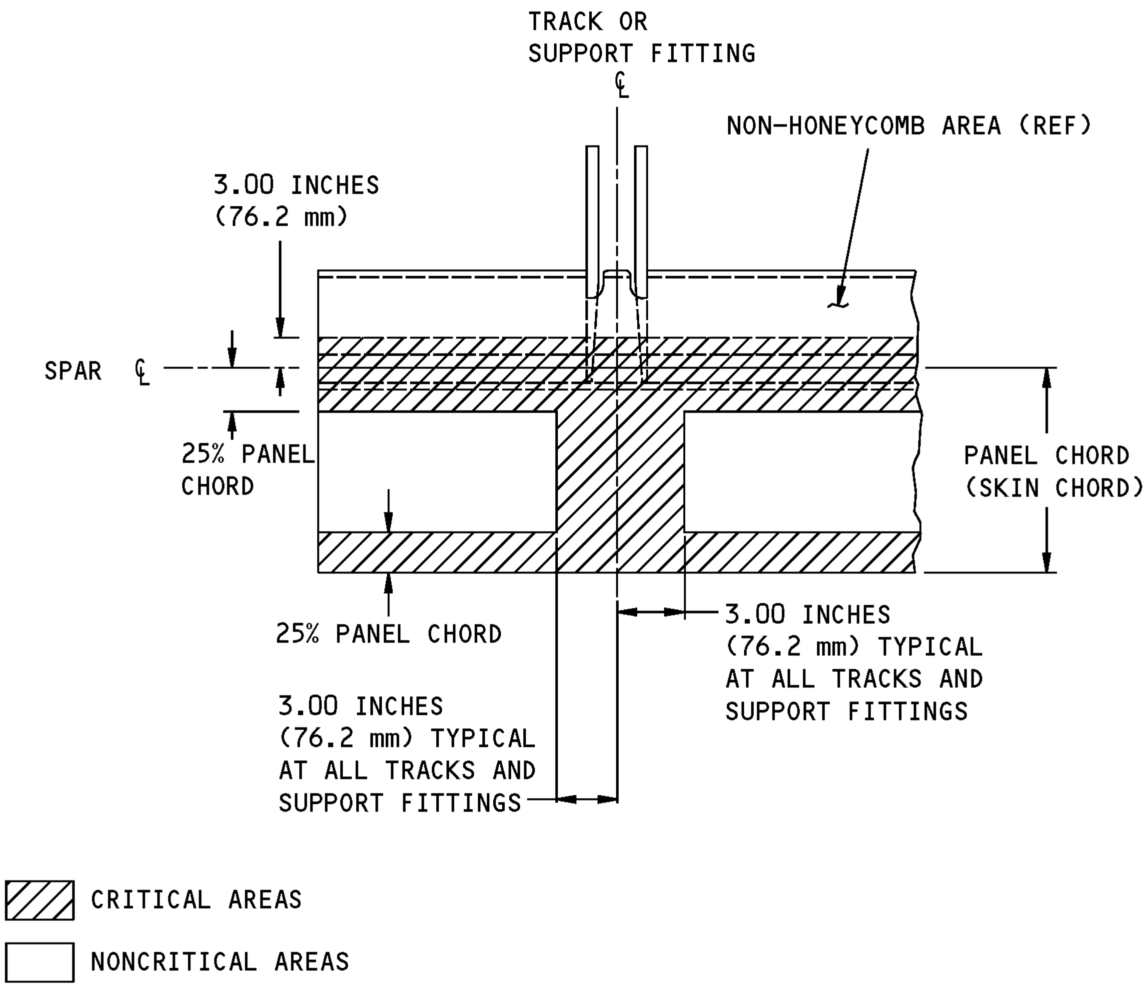

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 5)**




DENT THAT IS PERMITTED

F

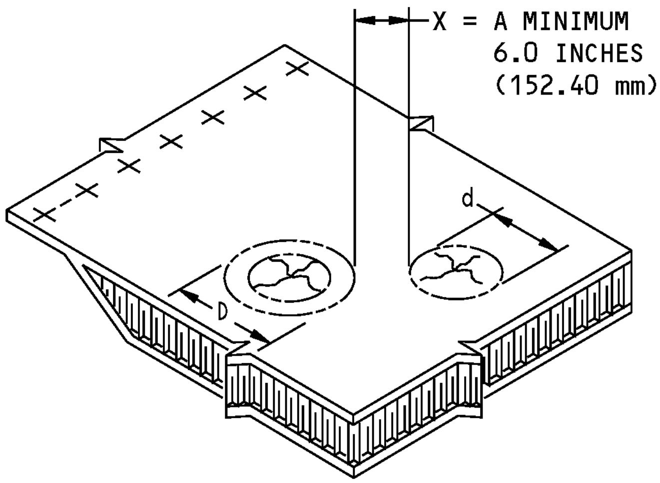
**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 5)**




TRAILING EDGE WEDGE SURFACES (ALUMINUM HONEYCOMB)

G

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 5)**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

D = THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

d = THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X = THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

DAMAGE THAT IS PERMITTED ON THE SURFACE OF
AN ALUMINUM HONEYCOMB PANEL

H

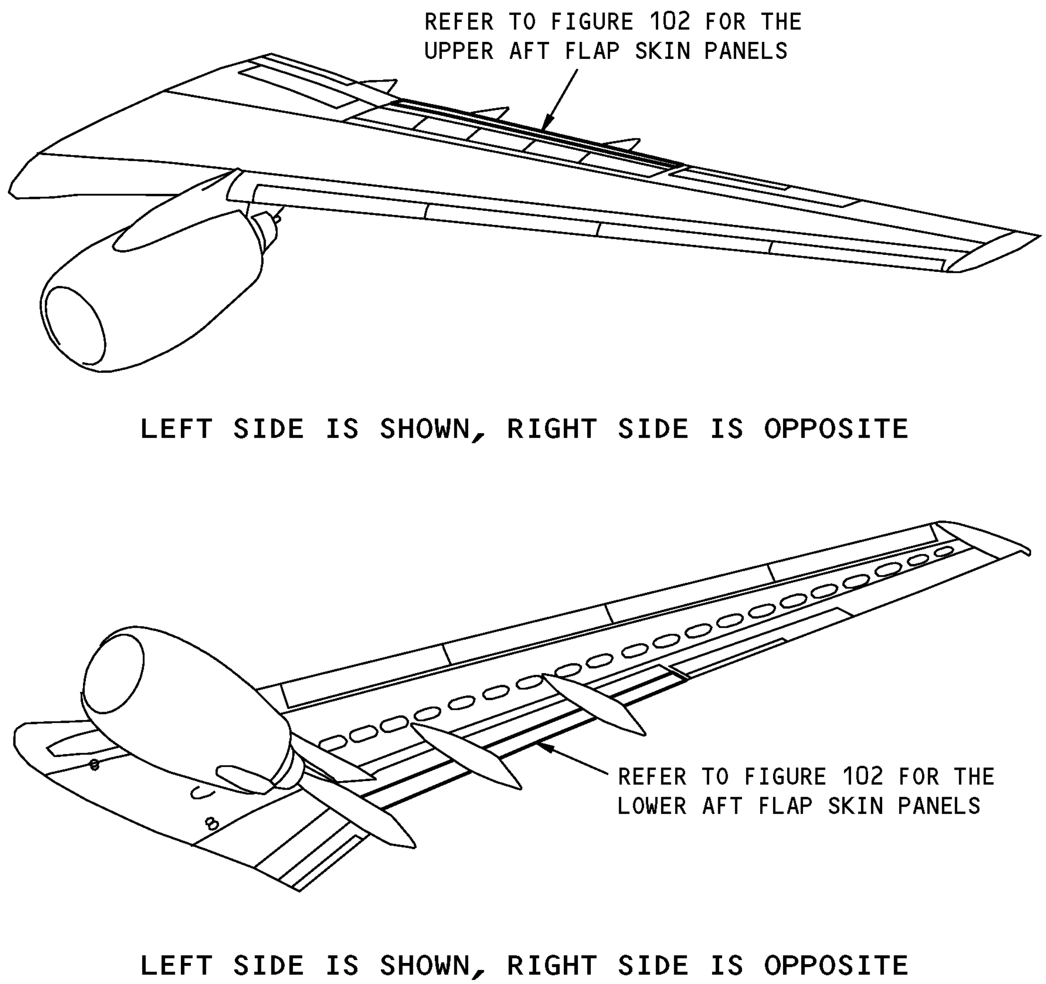
**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 5)**

## ALLOWABLE DAMAGE 5 - WING OUTBOARD TRAILING EDGE AFT FLAP SKIN

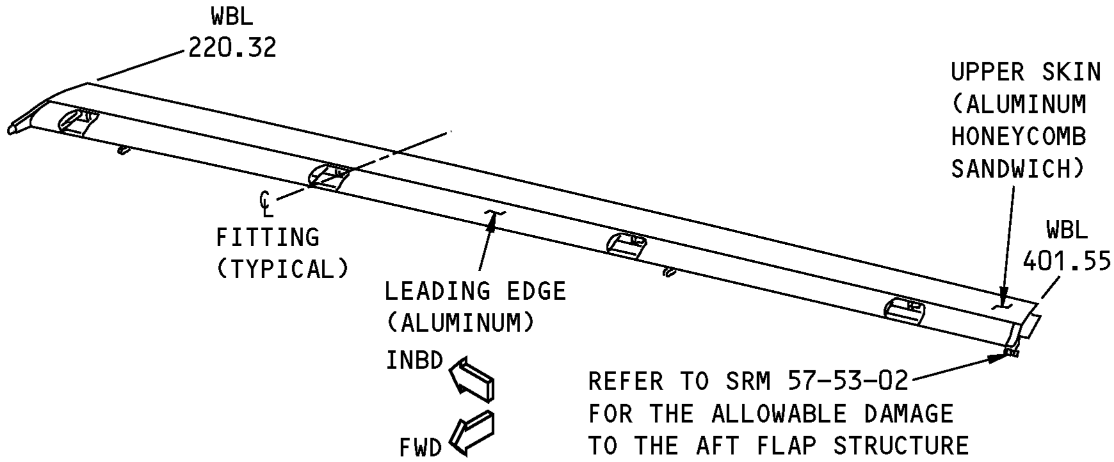
### 1. Applicability

A. This subject gives the allowable damage limits for the outboard trailing edge aft flap as shown in Figure 101.

B. Allowable Damage 5 is applicable to wing outboard trailing edge aft flaps which have initial 350 degree F (177 degree C) cure trailing edge wedge assemblies. Refer to drawing 113A3700-23/-24 for the 350 degree F (177 degree C) cure trailing edge aft flap assembly details.

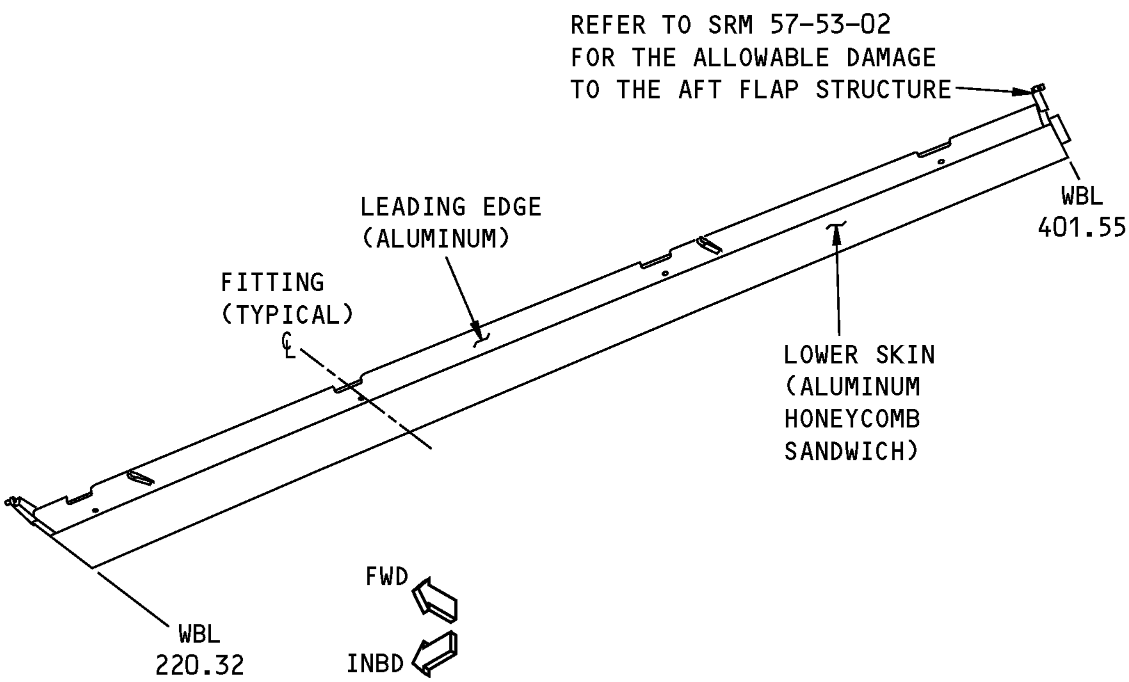



**Wing Trailing Edge Aft Flap Skin Panel Location**
**Figure 101**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANEL




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL

**Wing Outboard Trailing Edge - Aft Flap Skin Panel Location**
**Figure 102**

## 2. General

A. Refer to Paragraph 4 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(2) Apply a layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-01, GENERAL | Protective Treatment of Metallic and Composite Materials |
| 51-20-05, GENERAL | Repair Sealing |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-03, GENERAL | Sources for Non-Metallic Repair Materials |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-30-05, GENERAL | Equipment and Tools For Repairs |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-10 | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| 51-70-10, REPAIR P/B REPAIR | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Remove the damage as shown in Figure 103, Details A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103 Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Figure 103, Detail F .

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

b) The distance away from other damage is less than 0.50D ( D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted if:

1) It is a maximum of 0.25 inch in diameter and located in a non-critical area as shown in Figure 103, Detail G .

2) You do a close visual inspection of the damage to verify that there are no cracks or sharp edges.

3) It is a minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

4) You make a temporary seal with aluminum foil tape (speed tape).

**NOTE**: Use of vacuum and heat to remove moisture, is recommended.

a) Keep a record of the location.

b) Make sure the tape is in satisfactory condition and inspect at each 150 flight hour interval.

5) You do a permanent repair before 5,000 flight hours have occurred, as specified in 51-70-10.

(b) Holes and Punctures are not permitted in critical areas as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail G .

(5) Delamination is not permitted.

B. Leading Edge Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Figure 103, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Figure 103, Detail F .

1) If the damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

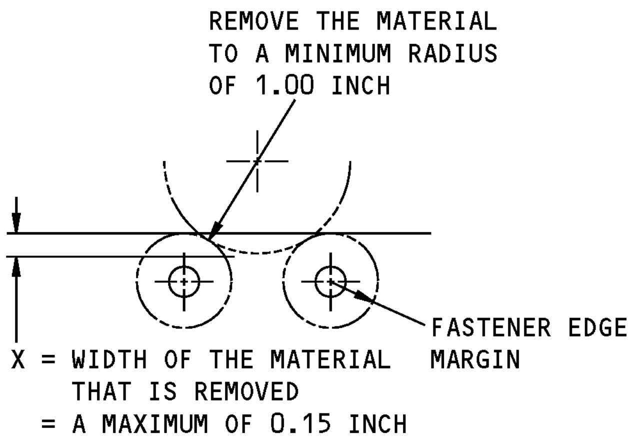
b) The distance away from other damage is less than 0.50D ( D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

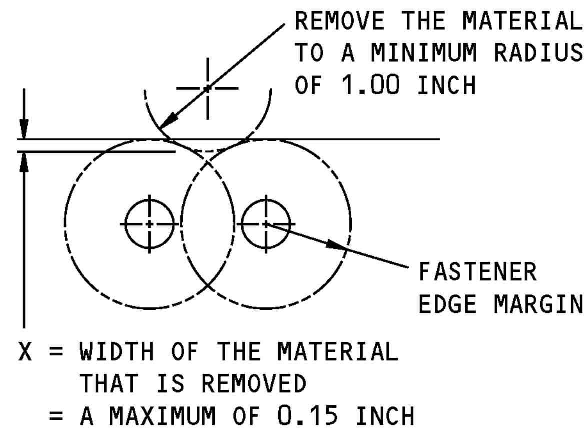
b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.



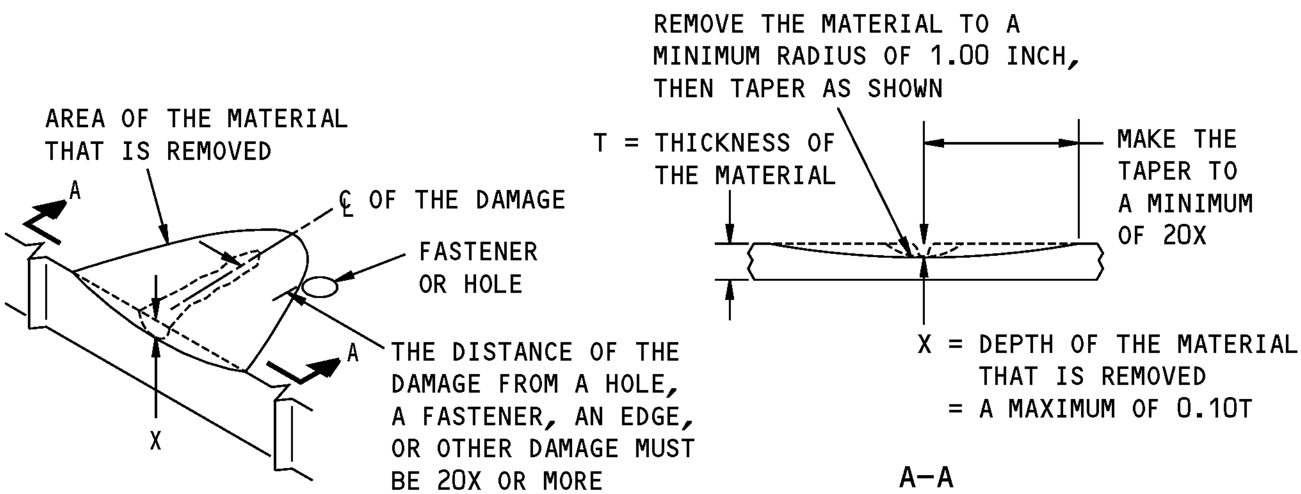

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

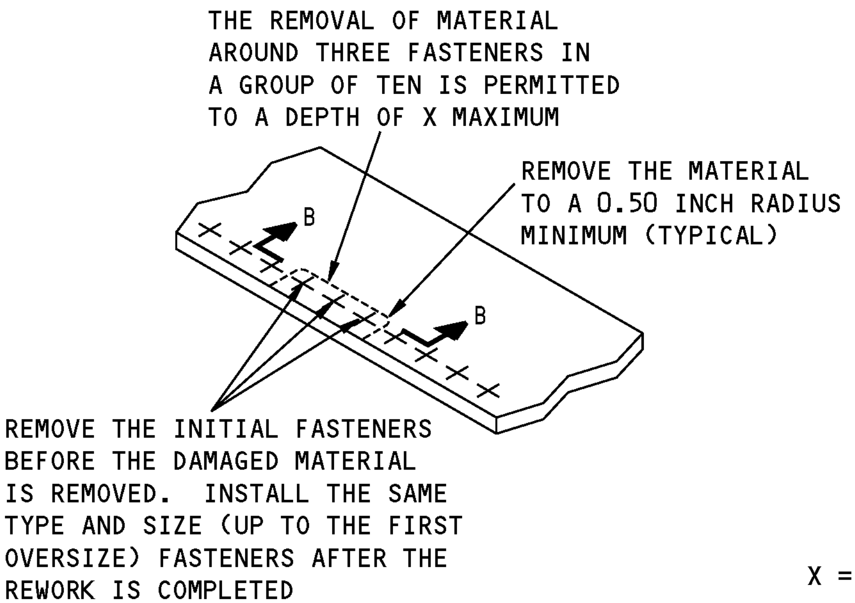
B




REMOVAL OF DAMAGED MATERIAL ON A SURFACE

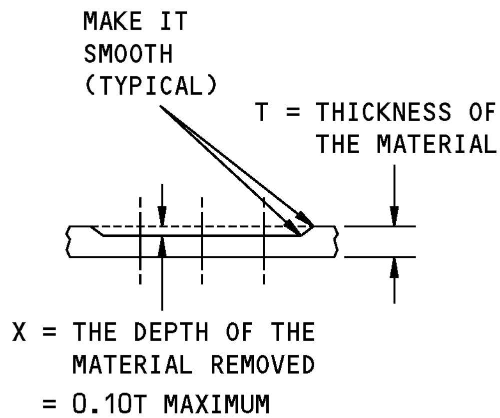
C

**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 4)**

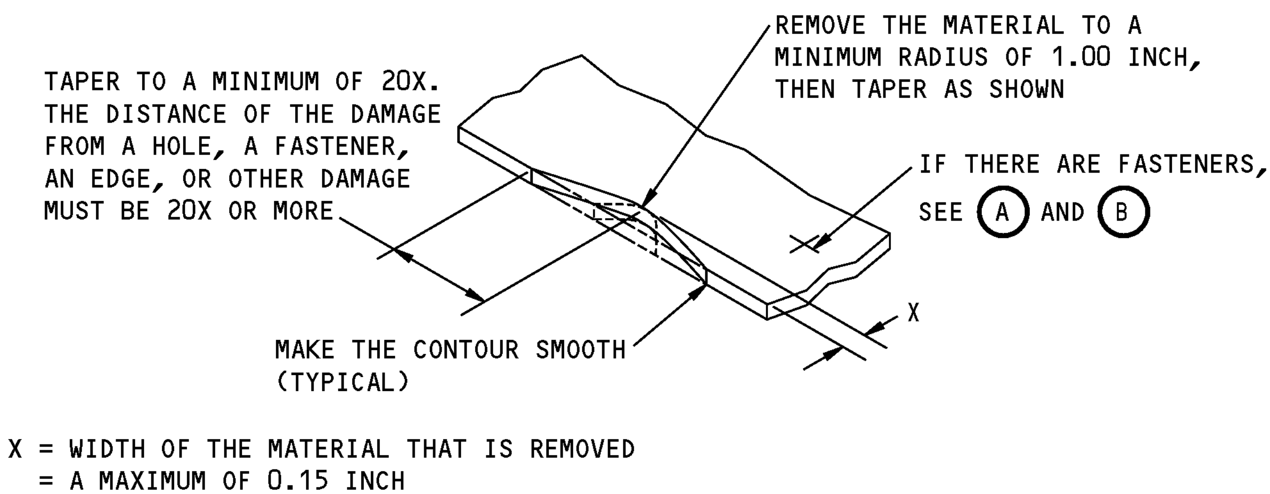



REMOVAL OF CORROSION AROUND THE FASTENERS

D



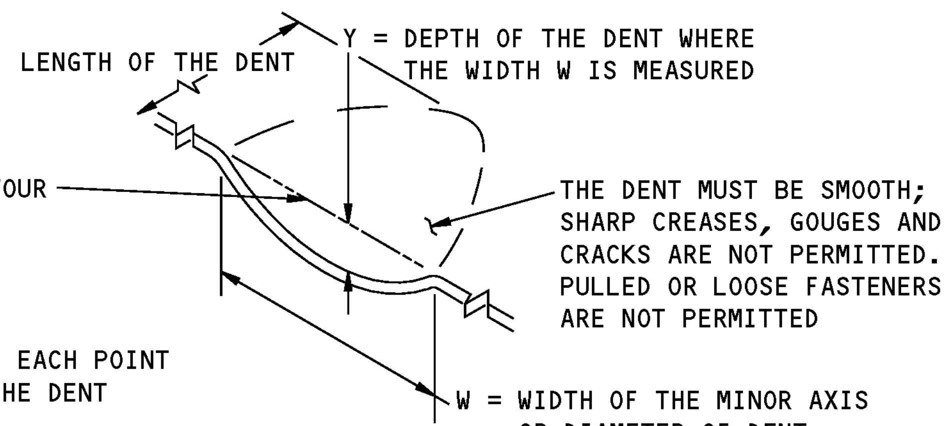

B-B




REMOVAL OF DAMAGED MATERIAL AT AN EDGE

E

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 4)**




$\frac{W}{Y}$ MUST BE 10 OR MORE AT EACH POINT ALONG THE LENGTH OF THE DENT

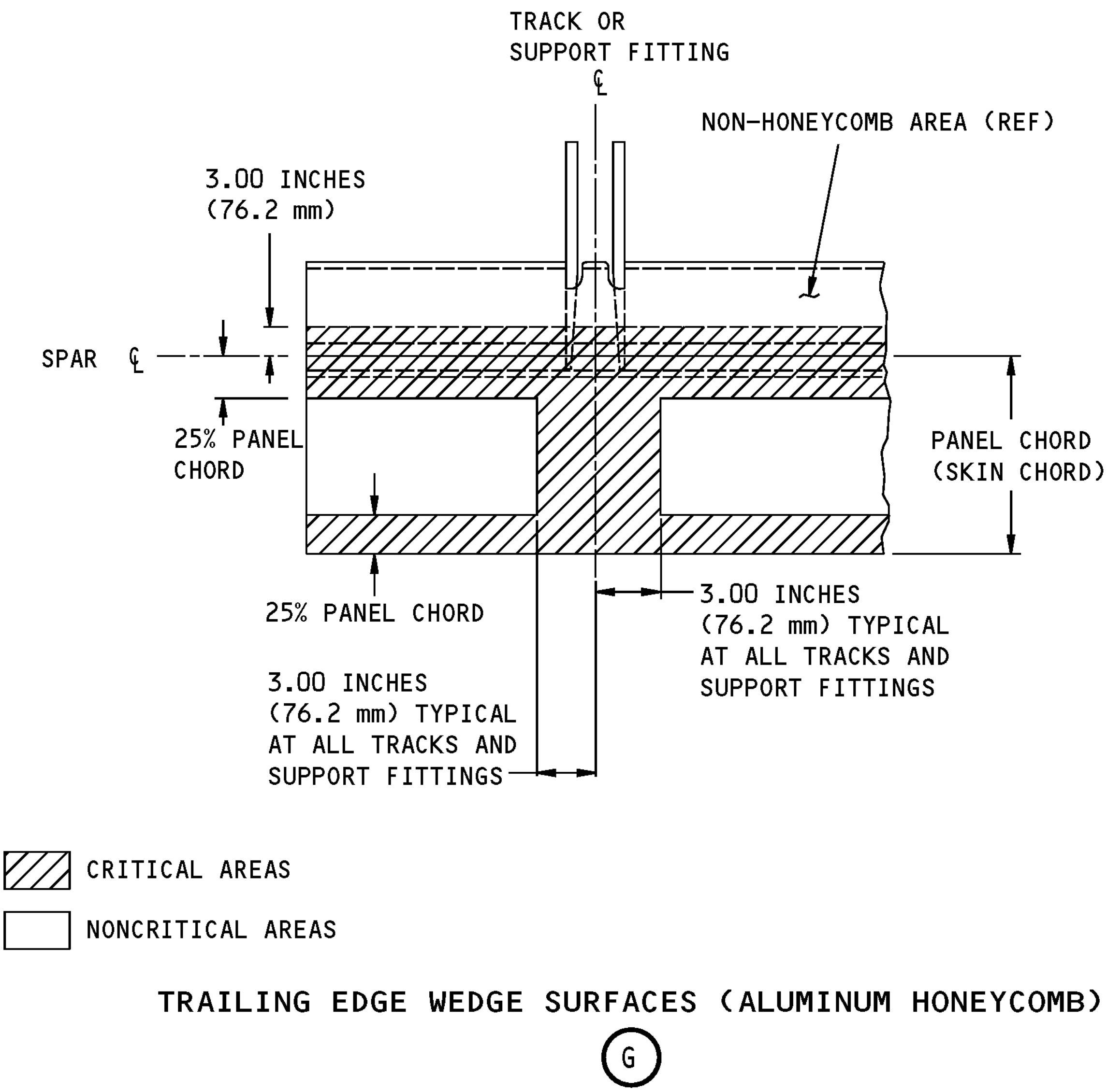
W = A MAXIMIUM OF 0.25 X (LENGTH OF THE PANEL ON THE SIDE OF THE PART WHICH HAS THE DAMAGE)

Y = A MAXIMUM OF 0.25 INCH OR 0.25 X (THE DEPTH OF THE HONEYCOMB CORE), THAT WHICH IS LESS

DENT THAT IS PERMITTED

F

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 4)**




**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 4)**

## REPAIR 1 - WING TRAILING EDGE - MAIN FLAP LOWER SKIN

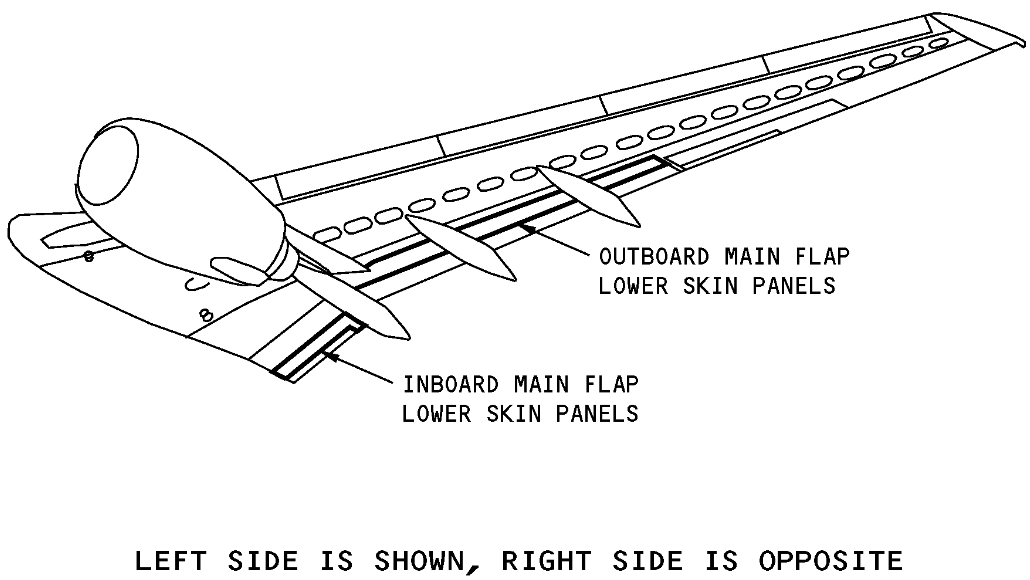
### 1. Applicability

A. Repair 1 is applicable to damage to the Inboard and Outboard Main Flap Lower Skins of the Wing Trailing Edge if they are:

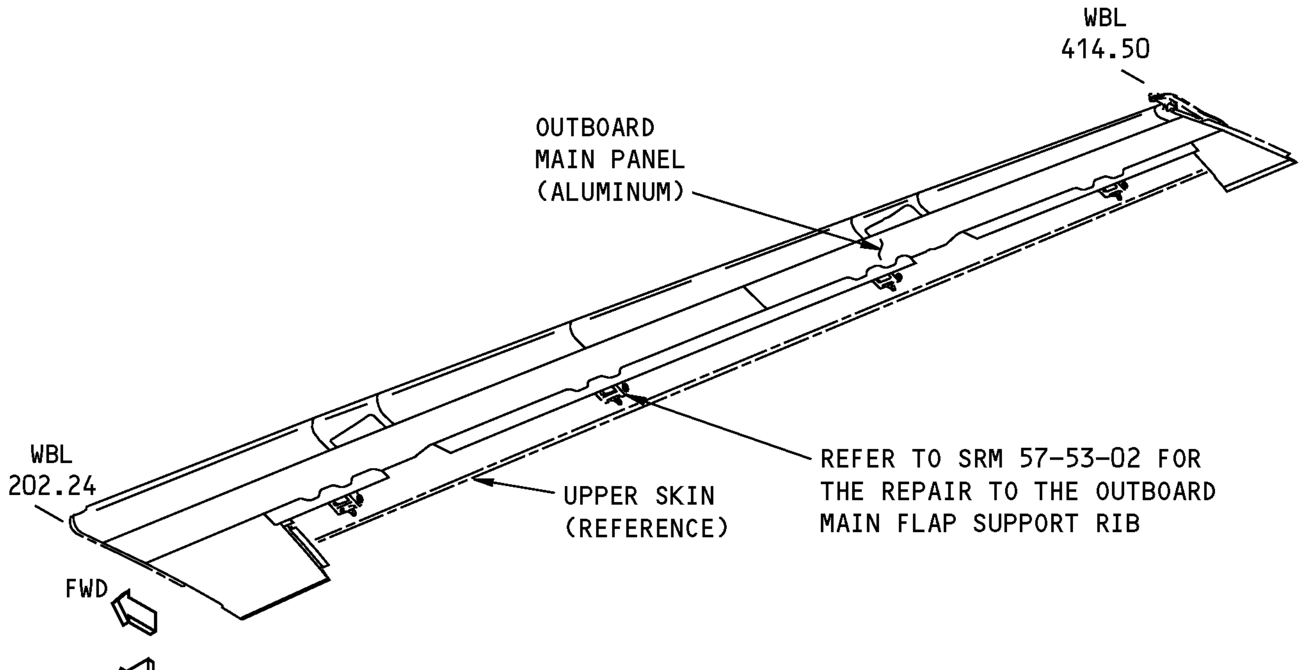
(1) Between the front and rear spars

(2) A maximum dimension of 3.0 inches.

B. The wing trailing edge main flaps are shown in Wing Trailing Edge - Location of Main Flap Skin Panels Made of Aluminum , Figure 201/REPAIR 1.

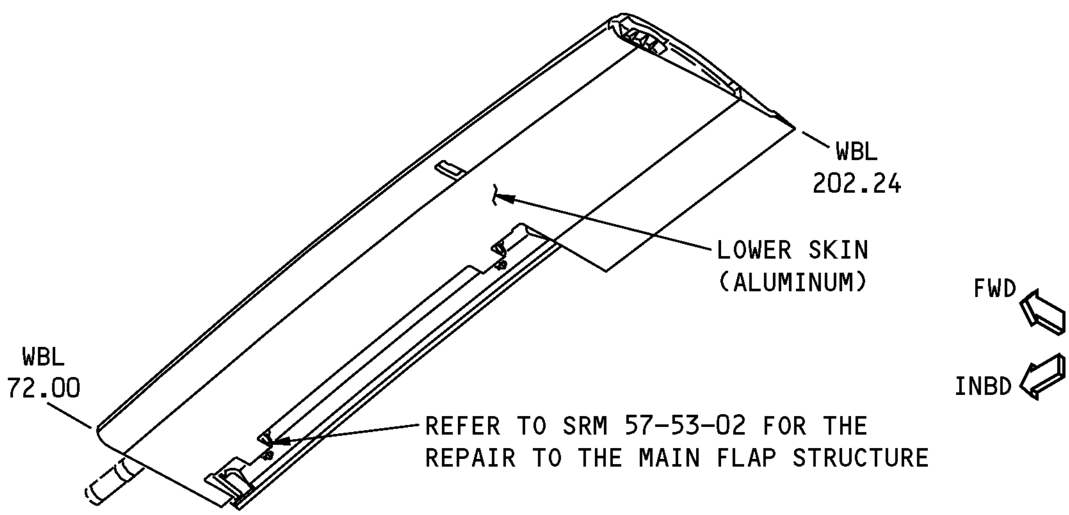



**Wing Trailing Edge - Location of Main Flap Skin Panels Made of Aluminum**
**Figure 201**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD MAIN FLAP LOWER SKIN PANEL




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

INBOARD MAIN FLAP LOWER SKIN PANEL

**Wing Trailing Edge - Main Flap Skin Panel**
**Figure 202**

## 2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 27-51-11 P/B 401 | INBOARD TRAILING EDGE FLAP - REMOVAL/INSTALLATION |
| AMM 27-51-21 P/B 401 | OUTBOARD TRAILING EDGE FLAP - REMOVAL/INSTALLATION |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Repair Instructions

A. Remove the main flap if necessary.
   - (1) Refer to AMM 27-51-11/401 for the inboard main flap.
   - (2) Refer to AMM 27-51-21/401 for the outboard main flap.

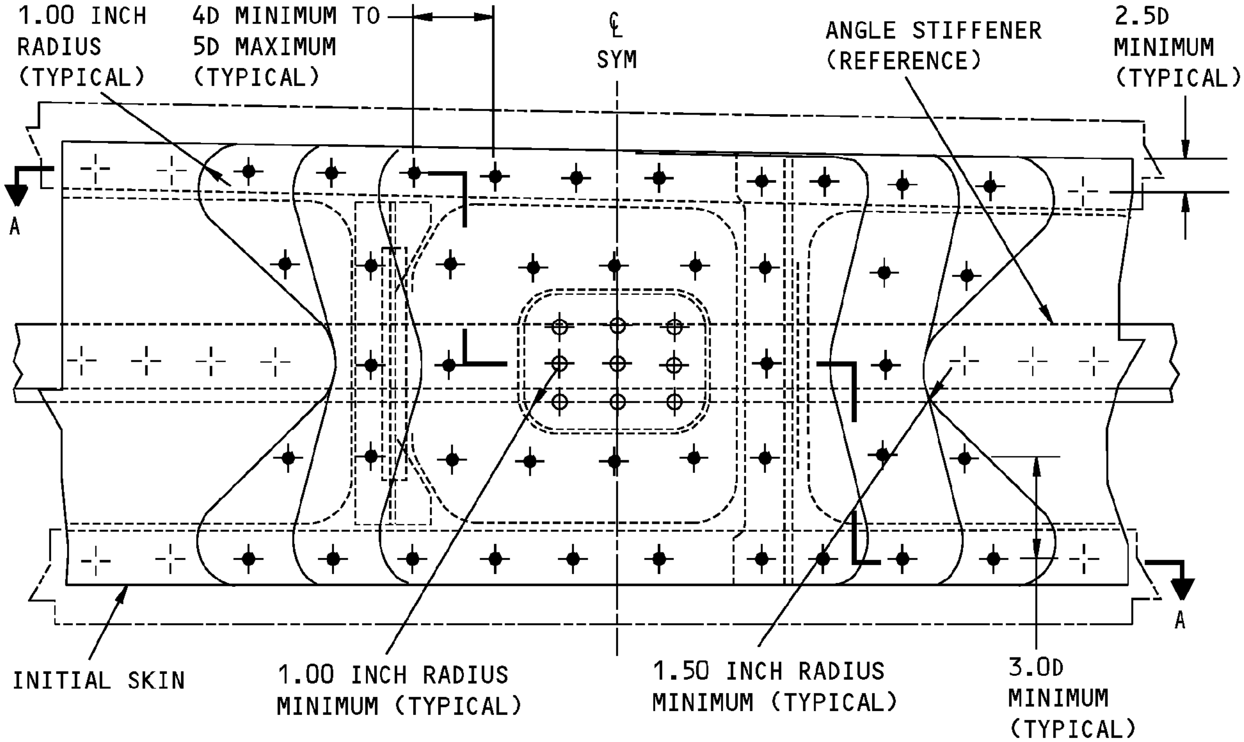
B. Remove the damage.
   - (1) If there is a crack, do the steps that follow:
     - (a) Cut and remove the damage. Refer to 51-10-02.
     - (b) Refer to Wing Trailing Edge Main Flap Lower Skin Repair, Figure 203/REPAIR 1 for the shape of the cut out.
     - (c) Make sure there is a minimum of two rows of repair fasteners around the edges of the cut out.

C. Make the repair parts as shown in Wing Trailing Edge Main Flap Lower Skin Repair, Figure 203/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair material.
   - (1) Refer to 51-40-08 for the countersink repair washers.
   - (2) Make the contour of the repair parts the same as the initial contour of the skin.

Table 201:

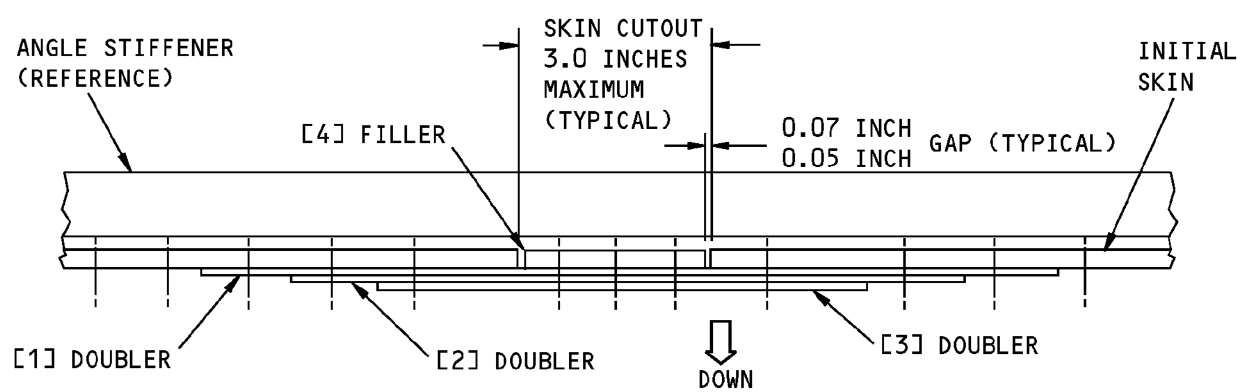
| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Doubler | 1 | Use bare or clad 2024-T3 that is 0.040 inch thick |
| [2] | Doubler | 1 | Use bare or clad 2024-T3 that is 0.040 inch thick |
| [3] | Doubler | 1 | Use bare or clad 2024-T3 that is 0.040 inch thick |
| [4] | Doubler | 1 | Use bare or clad 2024-T3 that is 0.040 inch thick |
| [5] | Filler | 1 | Use bare or clad 2024-T3 that is the same thickness as the initial skin |

D. Assemble the repair parts and drill the fastener holes.
   (1) Countersink the fastener holes in the repair parts to a depth that is 80 percent or less of the part thickness.
   **NOTE**: If the depth of the countersink is more than 80 percent of the part thickness a knife edge condition will result.
   (2) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.
E. Remove the repair parts.
F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts.
G. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the skin.
H. Apply one layer of BMS 10-79, Type II primer to the repair parts and to the bare edges of the initial skin. Refer to 51-20-01.
I. Install countersink washers with BMS 5-95 sealant at the initial fastener locations as necessary. Refer to 51-40-08.
J. Install the repair parts.
   (1) Apply BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.
K. Install the fasteners.
   (1) Install the hex drive bolts wet with BMS 5-95 sealant.
   (2) Install the rivets without sealant.
L. Install the skin panel on the main flap.
M. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.




NOTE: D=THE DIAMETER OF THE FASTENER

BOTTOM VIEW OF LOWER SKIN




A-A

FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION

REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX DRIVE BOLT WITH A BACC30M COLLAR.

REPAIR FASTENER LOCATION. INSTALL A BACR15CE6D RIVET.

**Wing Trailing Edge Main Flap Lower Skin Repair**
**Figure 203**

## REPAIR 2 - LEADING EDGE SKIN OF A WING TRAILING EDGE AFT FLAP

### 1. Applicability

A. Repair 2 is applicable to damage to wing inboard and outboard trailing edges in the areas that satisfy each of these two conditions:

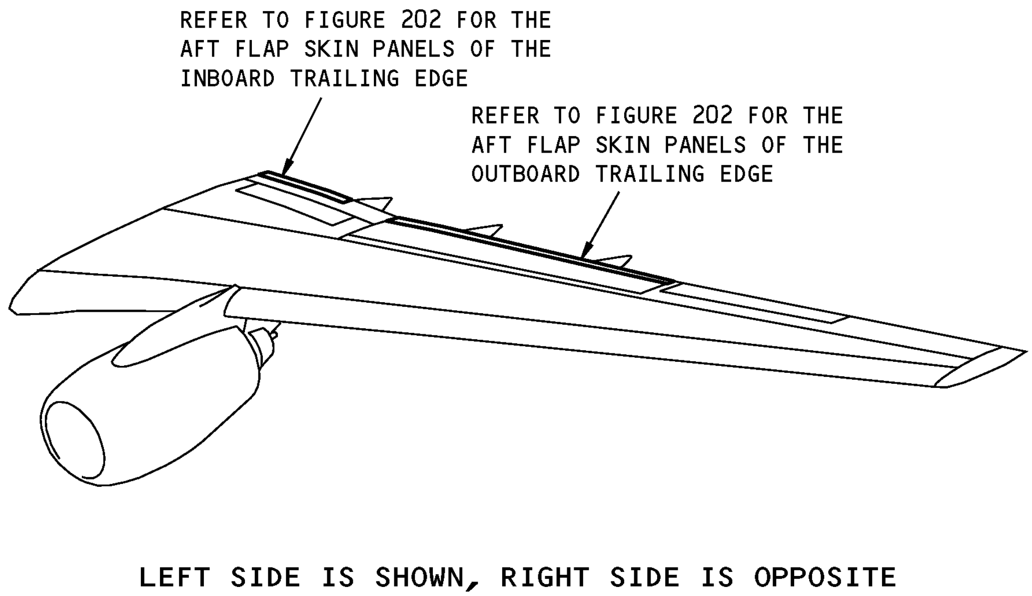
(1) The area is the aft flap leading edge shown in Wing Trailing Edge Aft Flap Skin Panel Locations, Figure 201/REPAIR 2.

(2) The area is between spar chords and ribs where the skin is of a constant thickness.

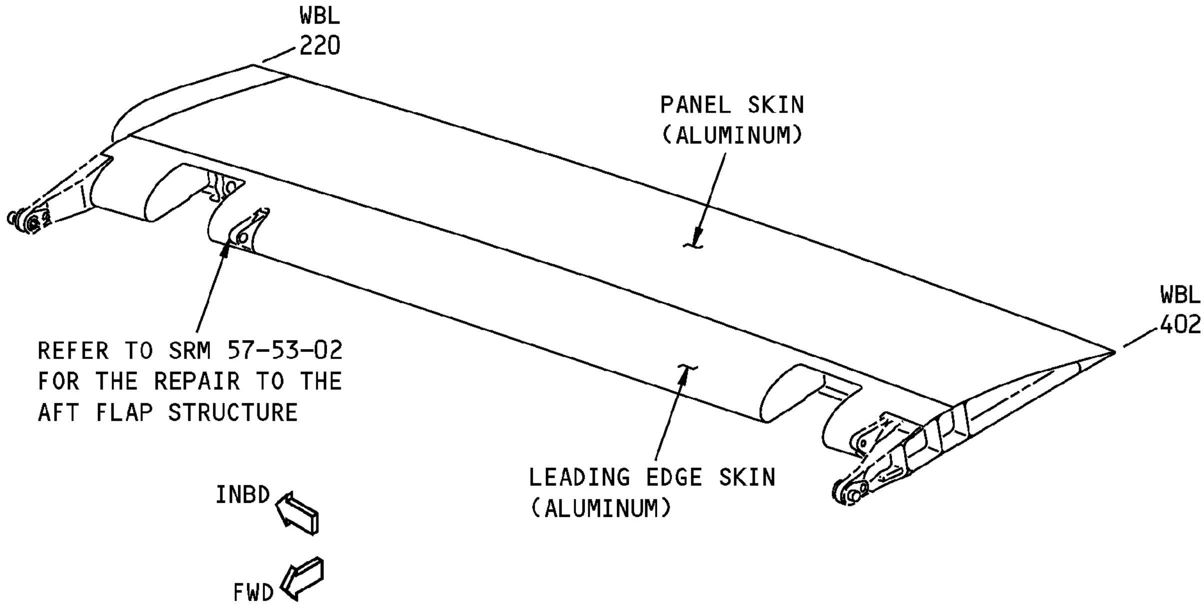
### 2. General

A. Repair 2 is a Category B repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

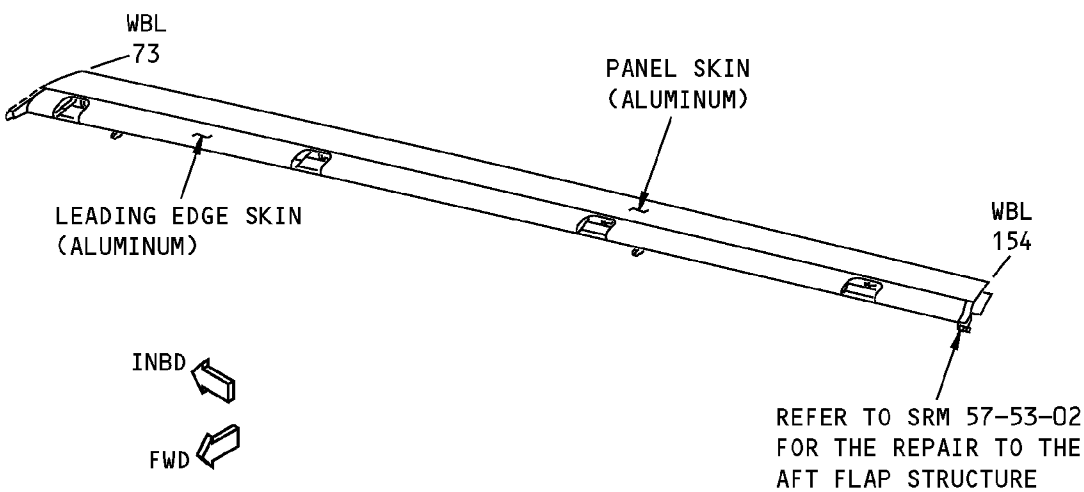



**Wing Trailing Edge Aft Flap Skin Panel Locations**
**Figure 201**




LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

INBOARD TRAILING EDGE AFT FLAP SKIN PANELS




LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

OUTBOARD TRAILING EDGE AFT FLAP SKIN PANELS

**Wing Trailing Edge - Aft Flap Leading Edge Skin Repair**
**Figure 202**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| AMM 27-51-12 P/B 401 | INBOARD AFT FLAP - REMOVAL/INSTALLATION |
| AMM 27-51-22 P/B 401 | OUTBOARD AFT FLAP - REMOVAL/INSTALLATION |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Repair Instructions

A. Cut and remove the damaged part of the leading edge skin as shown in Wing Trailing Edge - Aft Flap Leading Edge Skin Repair, Figure 203/REPAIR 2. Refer to 51-10-02 for the procedures to remove the damage.

(1) Make the cut in the shape of a rectangle with the longer sides parallel to the leading edge ribs.

(2) Make sure there is a minimum of two rows of repair fasteners around the edges of the cutout.

B. Put the skin that is around the damage back to the initial contour.

C. Make the repair parts as shown in Wing Trailing Edge - Aft Flap Leading Edge Skin Repair, Figure 203/REPAIR 2. Refer to Table 201/REPAIR 2 for the repair material.

(1) Make the contour of the repair parts the same as the contour of the initial skin.

**NOTE**: Make sure that the repair doublers do not overlap the chem-milled radius of the skin.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| **ITEM** | **PART** | **QUANTITY** | **MATERIAL** |
| [1] | Repair Plate | 1 | Use bare or clad 2024-T42 sheet that is 0.071 inch thick |
| [2] | Doubler | 1 | Use bare or clad 2024-T42 sheet that is 0.071 inch thick |
| [3] | Doubler | 1 | Use bare or clad 2024-T42 sheet that is 0.071 inch thick |
| [4] | Strap | 1 | Use bare or clad 2024-T4 sheet that is 0.071 inch thick |

D. Assemble the parts as shown in Wing Trailing Edge - Aft Flap Leading Edge Skin Repair, Figure 203/REPAIR 2.

E. Drill and countersink the fastener holes.

F. Remove the nicks, scratches, gouges, and sharp edges from the repair parts and the bare surfaces of the initial skin.

G. Apply a chemical conversion coating to the repair parts and bare surfaces of the initial skin. Refer to 51-20-01 for the chemical conversion coating procedures.

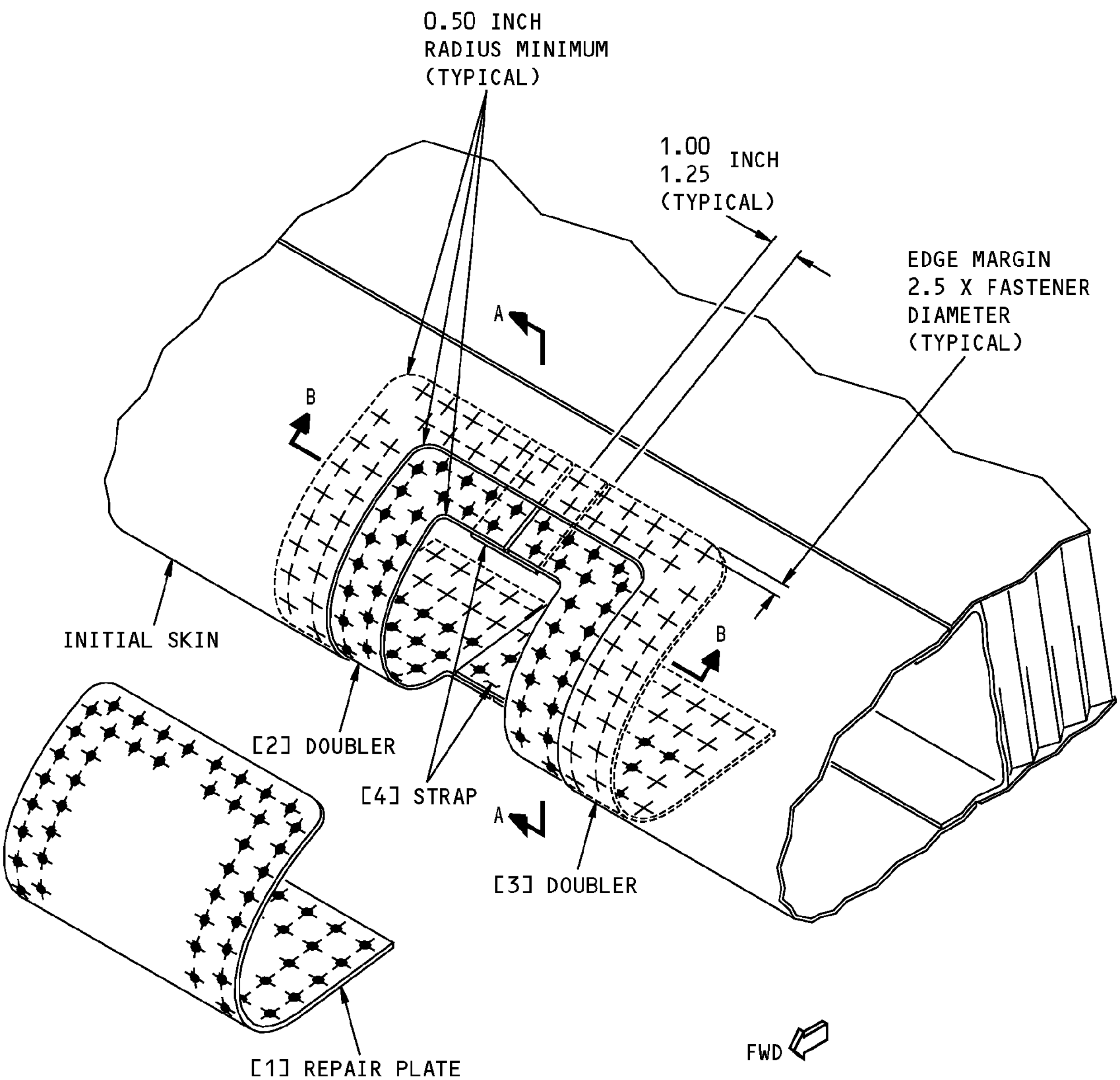
H. Apply one layer of BMS 10-11, Type I primer to the repair parts and the bare surfaces of the initial skin.

I. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05 for the procedures to apply the sealant.

J. Install the aluminum fasteners without sealant.

K. Fill the space between the [1] repair plate and the initial skin with BMS 5-95 sealant. Refer to 51-20-05.

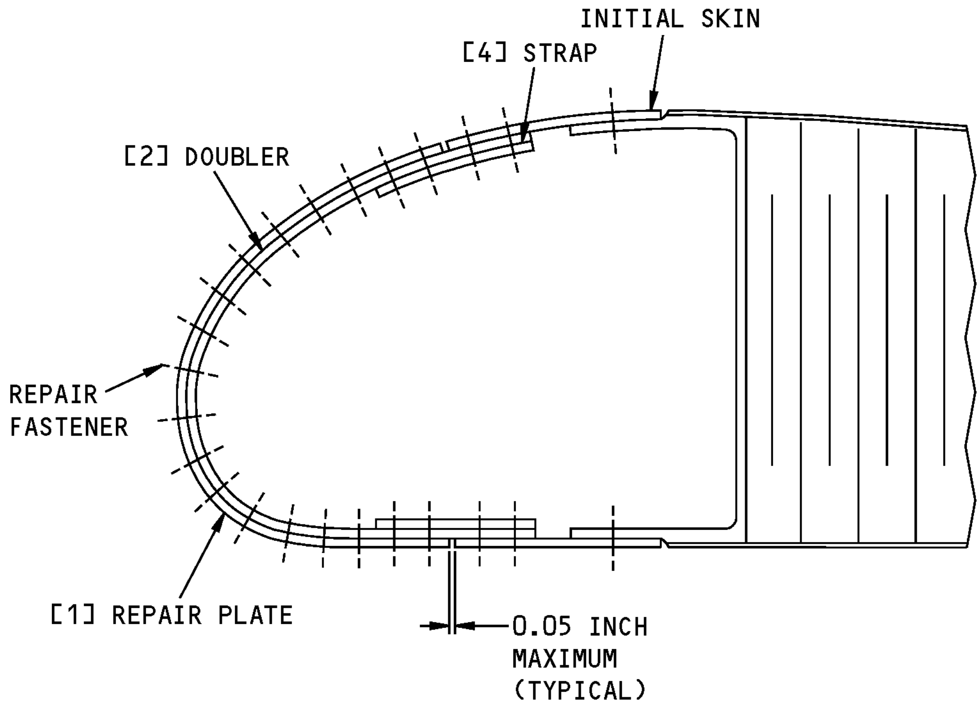
L. Apply the decorative finish to the repair area as given in AMM PAGEBLOCK 51-21-99/701.



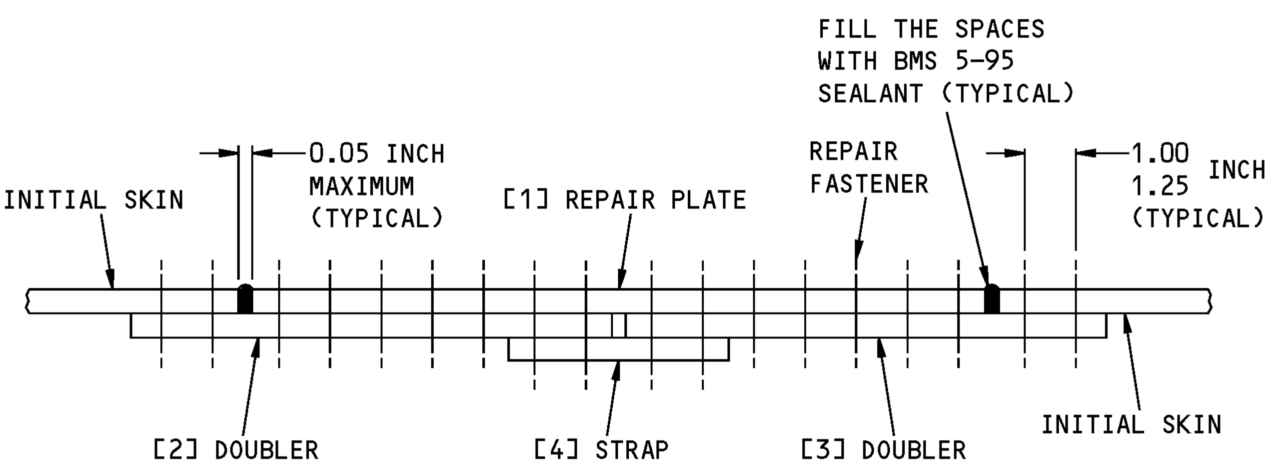

FASTENER SYMBOLS

- REPAIR FASTENER LOCATION. INSTALL A NAS1399D5 BLIND RIVET.
+ REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET.

**Wing Trailing Edge - Aft Flap Leading Edge Skin Repair**
**Figure 203 (Sheet 1 of 2)**




A-A



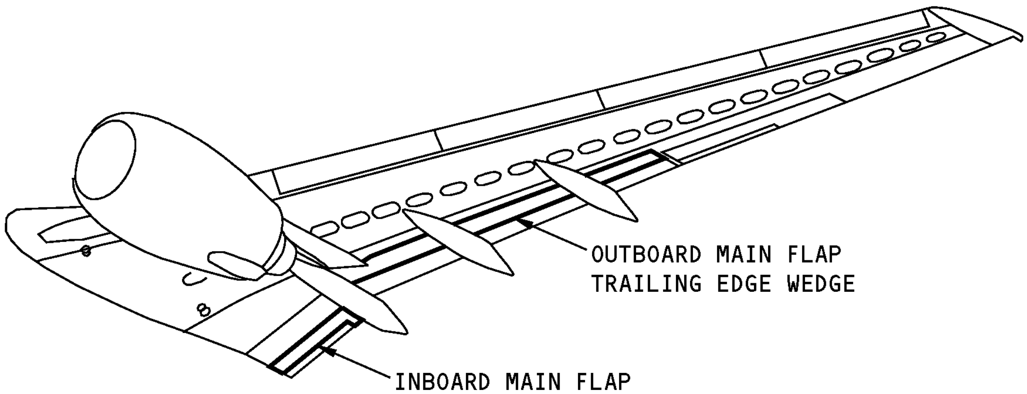

B-B

**Wing Trailing Edge - Aft Flap Leading Edge Skin Repair**
**Figure 203 (Sheet 2 of 2)**

5. Inspection Instructions

A. Do a detailed inspection of the repair at each 5000 flight hour interval or more frequently. Inspect all the blind rivets in the repair area. Blind rivets that are loose, missing, or damaged must be replaced.

## REPAIR 3 - INBOARD AND OUTBOARD TRAILING EDGE MAIN FLAP WEDGE

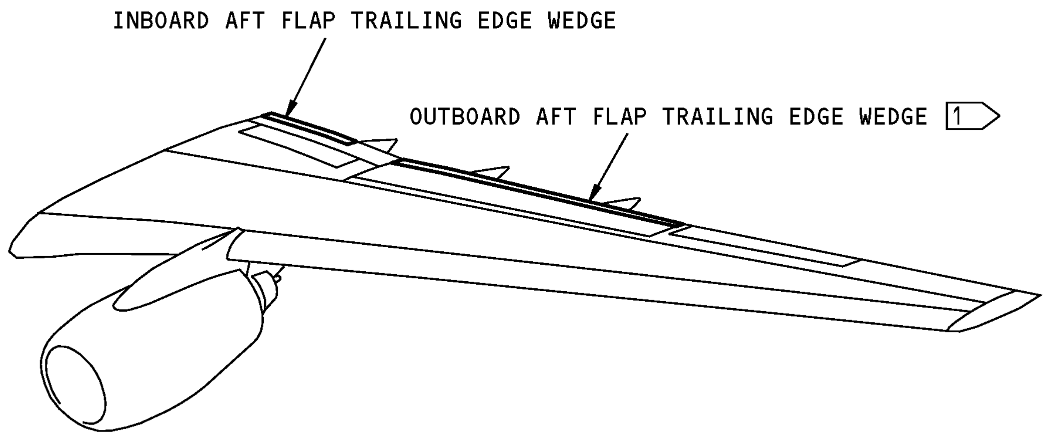



NOTE: BOEING HAS NOT FOUND IT NECESSARY TO SUPPLY REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Inboard and Outboard Main Flap Trailing Edge - Trailing Edge Wedge Locations**
**Figure 201**

## REPAIR 4 - INBOARD AND OUTBOARD AFT FLAP TRAILING EDGE WEDGE




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

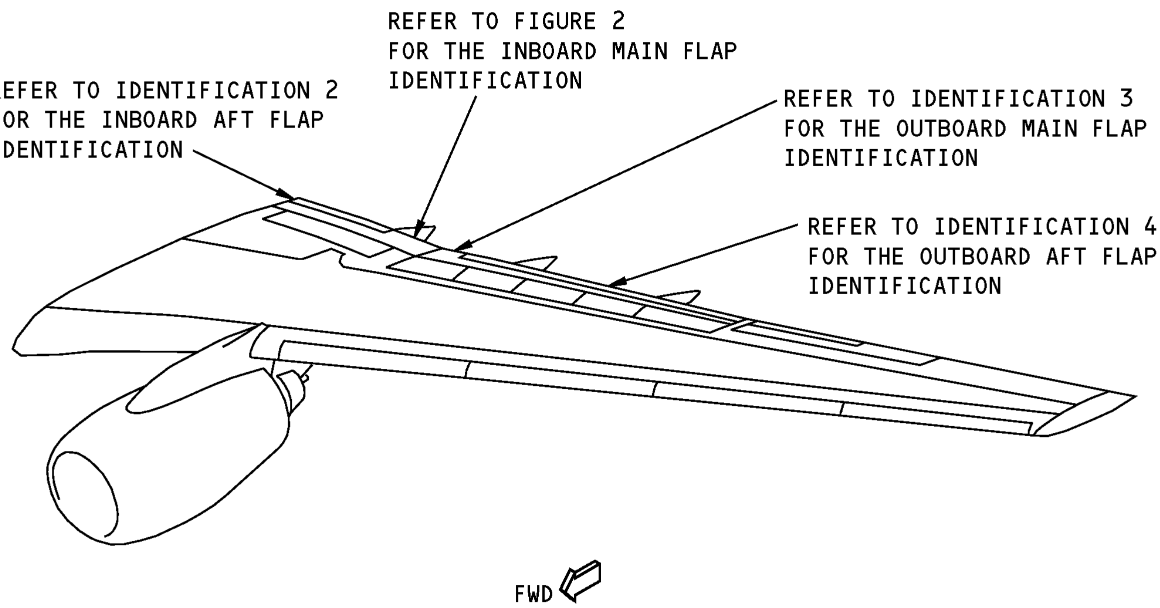
NOTES

- REFER TO SRM 51-70-10 FOR TYPICAL METAL REPAIRS FOR THE INBOARD AND OUTBOARD AFT FLAP TRAILING EDGE WEDGE. MAKE SURE THERE IS SUFFICIENT CLEARANCE FROM ADJACENT STRUCTURE TO INSTALL THE REPAIR PARTS.
- MAKE SURE THAT THE TOTAL LENGTH OF EXTERNAL REPAIRS TO BOTH THE TRAILING EDGE WEDGES IS NOT MORE THAN 24 INCHES IN THE INBOARD TO OUTBOARD DIRECTION. A REPAIR THAT INCLUDES BOTH THE UPPER AND LOWER SURFACES IS COUNTED ONLY ONCE IN THE TOTAL LENGTH.

1 ONLY ADHESIVES EA-9628 (DEXTER CORPORATION) OR AF-163 (3M AEROSPACE MATERIALS) ARE PERMITTED ON THE OUTBOARD AFT FLAP TRAILING EDGE WEDGES.

**Wing Inboard and Outboard Aft Flap - Trailing Edge Wedge Locations**
**Figure 201**

## IDENTIFICATION 1 - WING TRAILING EDGE INBOARD MAIN FLAP STRUCTURE



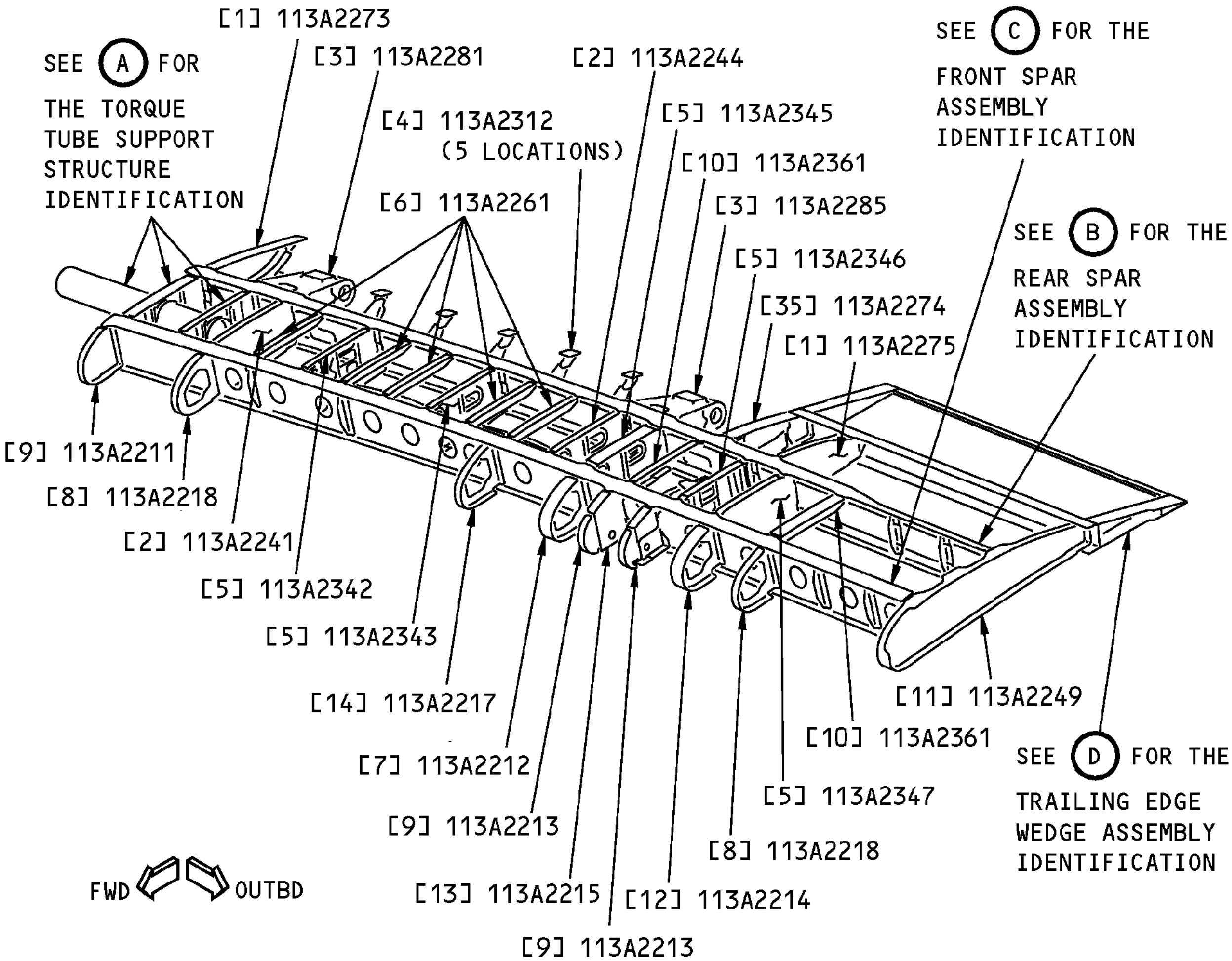

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Trailing Edge Inboard Main Flap Structure Locations**
**Figure 1**

**Table 1**:

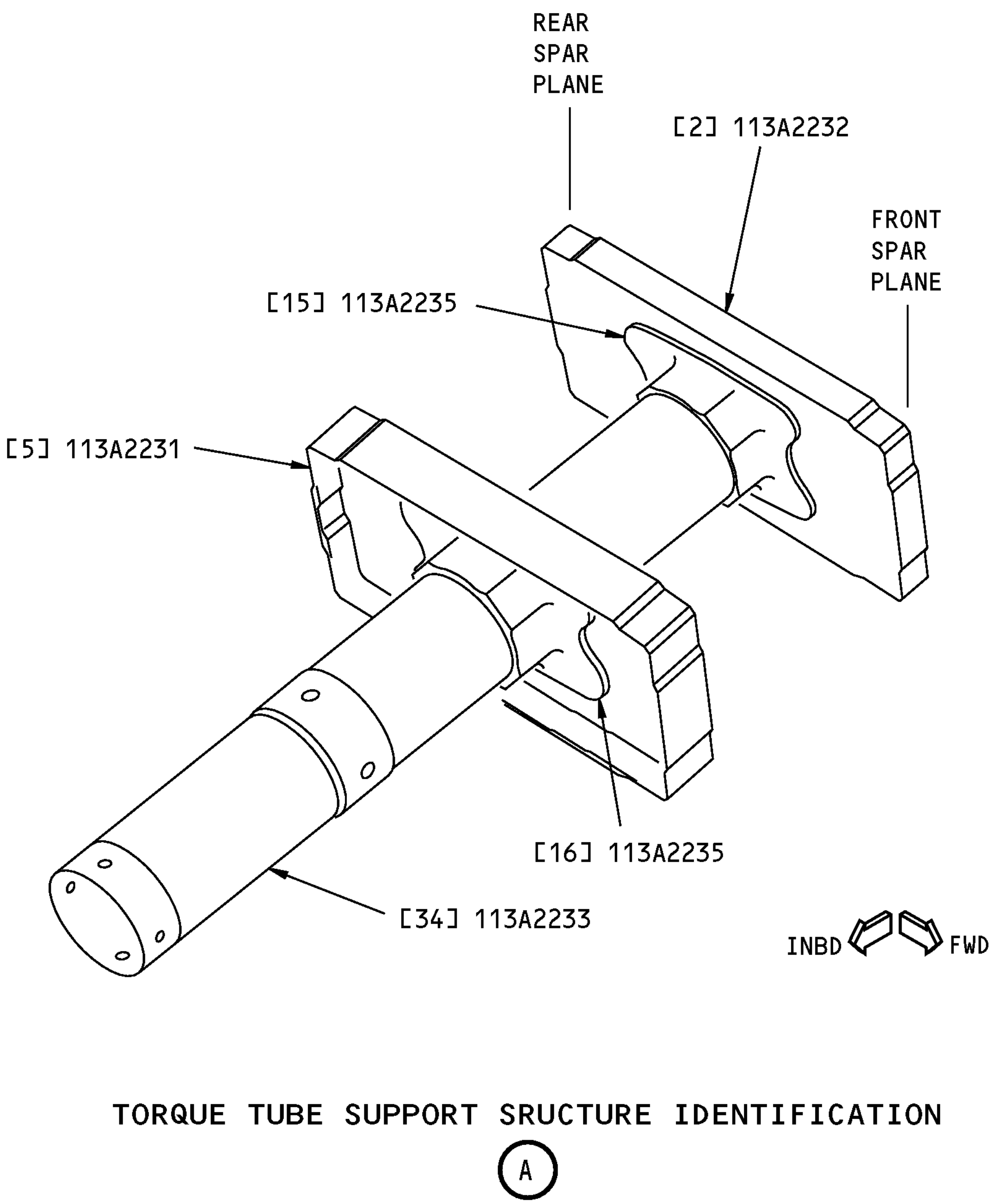
| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable T.E. Functional Product Collector |
| 113A2001 | Flap Installation - Inboard Trailing Edge Flap |
| 113A2002 | Flap Assembly - Inboard Trailing Edge Flap |
| 113A2100 | Main Flap Assembly - Inboard Trailing Edge Flap |
| 113A2107 | Fitting Assembly - Fairing Attach, Inboard Main Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A2110 | Front Spar Assembly - Inboard Main Flap |
| 113A2130 | Rear Spar Assembly - Inboard Main Flap |
| 113A2215 | Support Fitting Assembly - Crankshaft, Aft Flap Drive, Inboard Main Flap |
| 113A2251 | Fitting Assembly - Inboard Bellcrank Support, Inboard Main Flap |
| 113A2252 | Support Assembly - Outboard, Bellcrank Aft Flap Drive, Inboard Main Flap |
| 113A2280 | Support Assembly - Inboard, Aft Flap Track Support |
| 113A2285 | Support Assembly - Outboard, Aft Flap Track Support |
| 113A2500 | Wedge Assembly - Bonded Part, Inboard Main Flap |

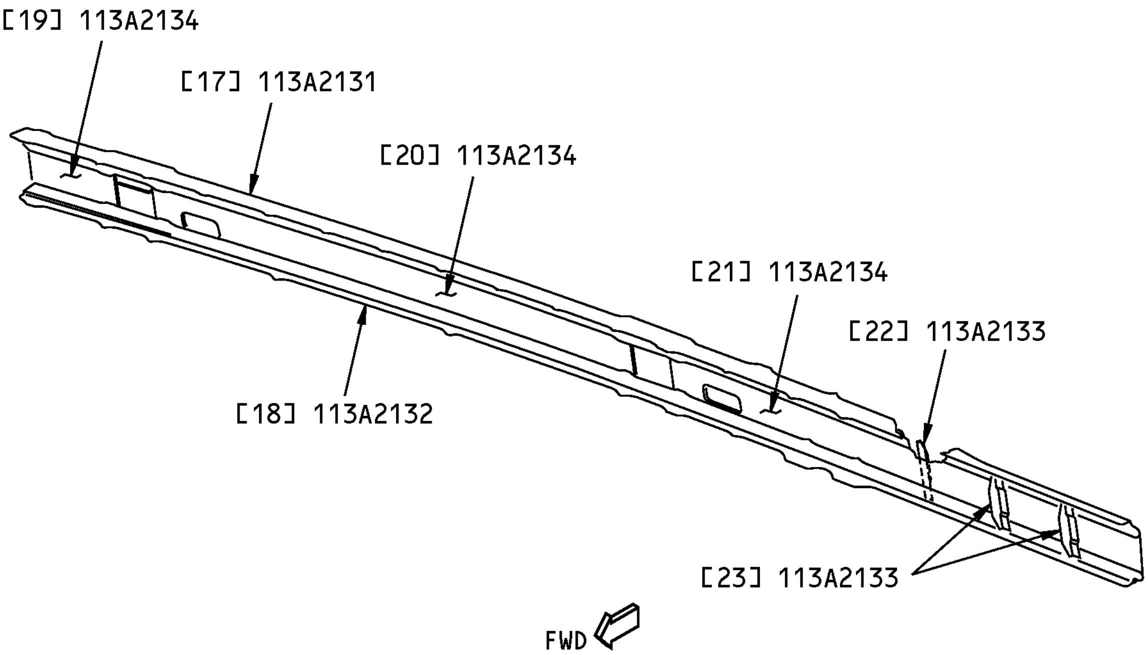



NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Wing Trailing Edge Inboard Main Flap Structure Identification**
**Figure 2 (Sheet 1 of 5)**

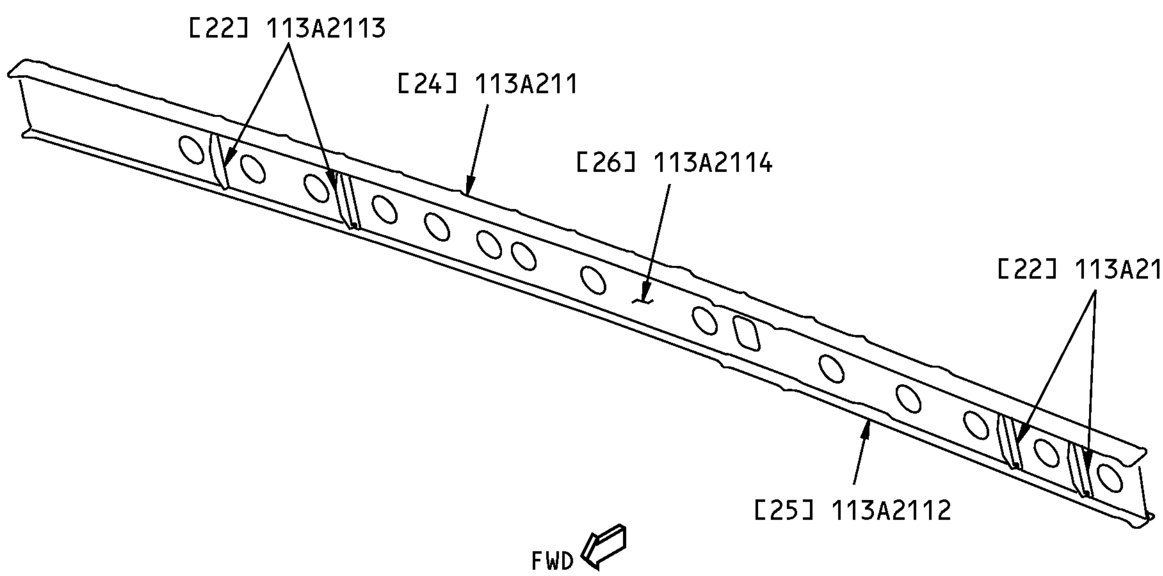



**Wing Trailing Edge Inboard Main Flap Structure Identification**
**Figure 2 (Sheet 2 of 5)**




REAR SPAR ASSEMBLY IDENTIFICATION

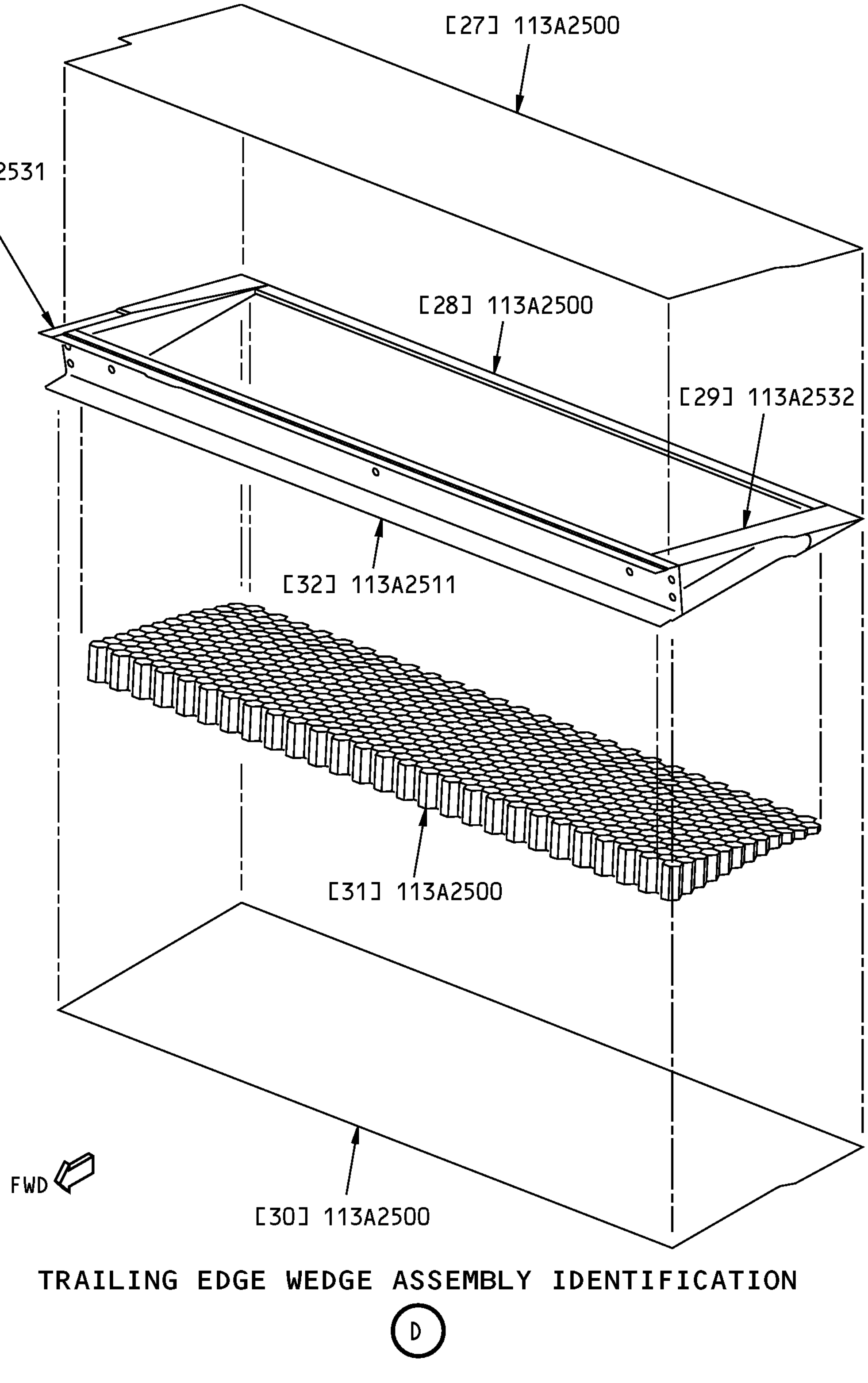
B



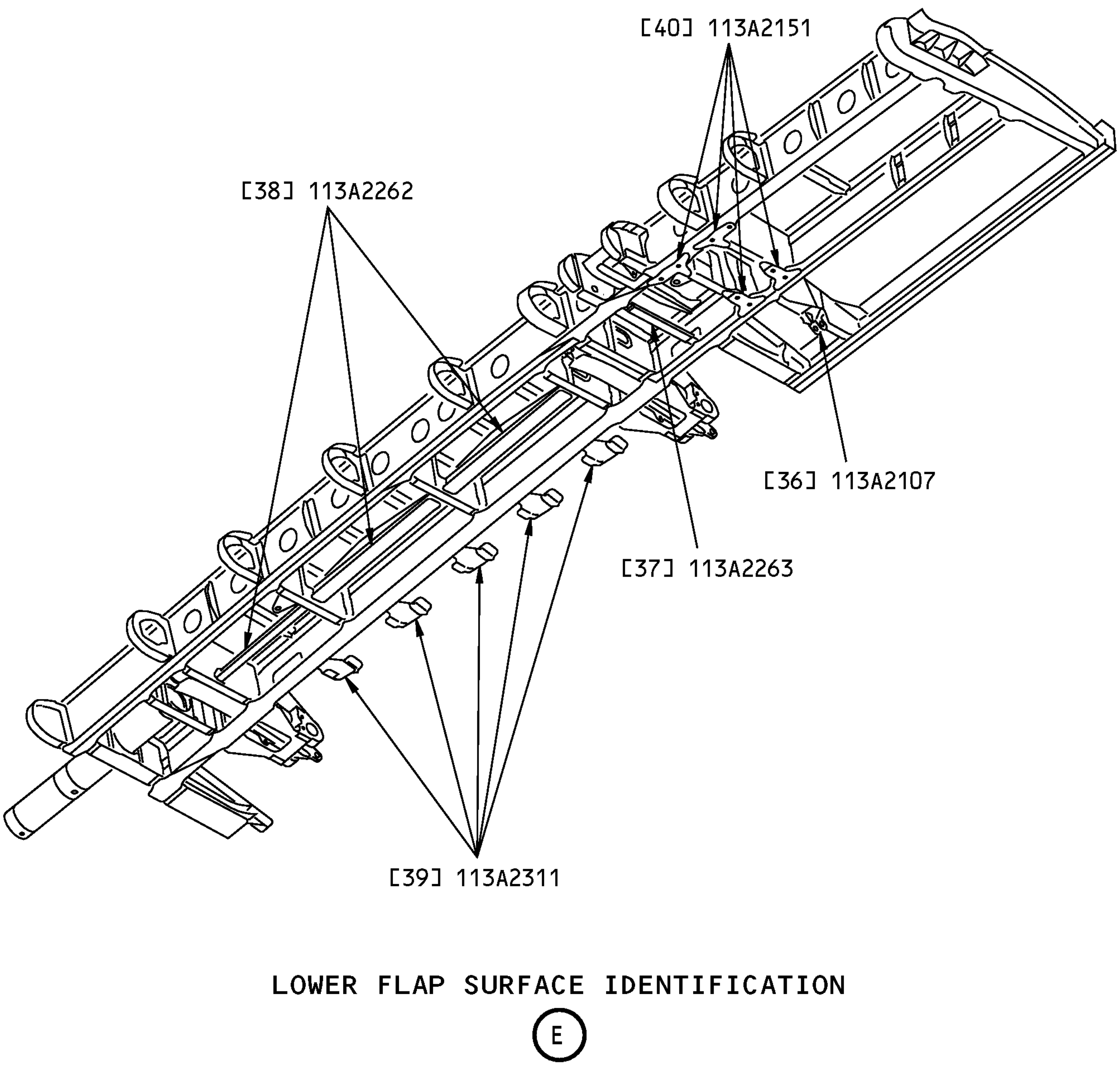

FRONT SPAR ASSEMBLY IDENTIFICATION

C

**Wing Trailing Edge Inboard Main Flap Structure Identification**
**Figure 2 (Sheet 3 of 5)**




**Wing Trailing Edge Inboard Main Flap Structure Identification**
**Figure 2 (Sheet 4 of 5)**




**Wing Trailing Edge Inboard Main Flap Structure Identification**
**Figure 2 (Sheet 5 of 5)**

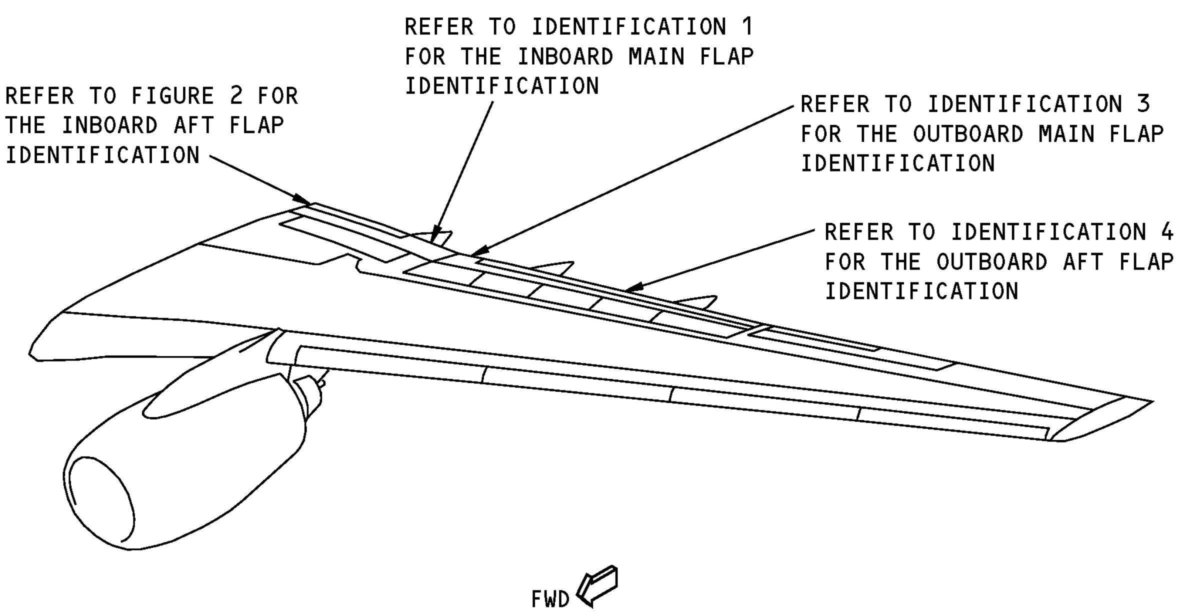
**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Rib - Wedge | | 7050-T7451 plate as given in AMS 4050 | |
| [2] | Rib - Inspar | | 7050-T7451 plate as given in AMS 4050 | |
| [3] | Fitting | | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Tee - Cove Support | | BAC1505-100967 2024-T8511 extrusion as given in QQ-A-200/3 | |
| [5] | Rib - Inspar | | 7050-T7451 plate as given in AMS 4050 | |
| [6] | Stiffener | | 7050-T7451 plate as given in AMS 4050 | |
| [7] | Rib - LE | | 7050-T7451 plate as given in AMS 4050 | |
| [8] | Rib - LE | | 7050-T7451 plate as given in AMS 4050 | |
| [9] | Rib - LE | 0.90 (22.86) | 7050-T7451 plate as given in AMS 4050 | |
| [10] | Stiffener | | 7050-T7451 plate as given in AMS 4050 | |
| [11] | Rib - Closure | | 7050-T7451 plate as given in AMS 4050 | |
| [12] | Rib - LE | | 7050-T7451 plate as given in AMS 4050 | |
| [13] | Support Fitting | | 7050-T7451 plate as given in AMS 4050 | |
| [14] | Rib - LE | | 7050-T7451 plate as given in AMS 4050 | |
| [15] | Fitting - Torque Tube | | Ti-6Al-4V plate as given in Mil-T-9046, Code AB-1, in the annealed condition | |
| [16] | Fitting - Torque Tube | | Ti-6Al-4V plate as given in Mil-T-9046, Code AB-1, in the annealed condition | |
| [17] | Chord - Upper | | 7050-T7451 plate as given in AMS 4050 | |
| [18] | Chord - Lower | | 7050-T7451 plate as given in AMS 4050 | |
| [19] | Web | 0.100 (2.54) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [20] | Web | 0.160 (0.41) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [21] | Web | 0.200 (0.51) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [22] | Stiffener | | BAC1503-100180 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [23] | Stiffener | 0.90 (22.86) | 7050-T7451 plate as given in AMS 4050 | |
| [24] | Chord - Upper | | 7050-T7451 plate as given in AMS 4050 | |
| [25] | Chord - Lower | | 7050-T7451 plate as given in BMS 7-323, Type III | |
| [26] | Web | 0.063 (1.60) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [27] | Skin - Upper | 0.016 (0.41) | 7075-T6 bare sheet as given in QQ-A-250/12 | |
| [28] | Filler | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 | |
| [29] | Rib - Closure | | 2024-T351 plate as given in QQ-A-250/4 | |
| [30] | Skin - Lower | 0.020 (0.51) | 7075-T6 bare sheet as given in QQ-A-250/12 | |
| [31] | Honeycomb Core | | Aluminum honeycomb core as given in BMS 4-4, Type 4-25, Class NPA, Form B, Grade I | |
| [32] | Spar - T.E. Wedge | | 2024-T351 plate as given in QQ-A-250/4 | |
| [33] | Rib - Closure | | 2024-T351 plate as given in QQ-A-250/4 | |

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [34] | Tube - Torque | | Ti-6Al-4V plate as given in Mil-T-9047 | |
| [35] | Rib - Wedge | | 7050-T7451 plate as given in AMS 4050 | |
| [36] | Fitting | | 7050-T7451 plate as given in AMS 4050 | |
| [37] | Stiffener | | BAC1503-100144 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [38] | Stiffener | | 7050-T7451 plate as given in AMS 4050 | |
| [39] | Rib - Cove Support | 0.063 (1.60) | 2024-T351 plate as given in QQ-A-250/4 | |
| [40] | Bolt Pad | 0.25 (6.35) | 7050-T7351 plate as given in QQ-A-250/12 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - WING TRAILING EDGE INBOARD AFT FLAP STRUCTURE



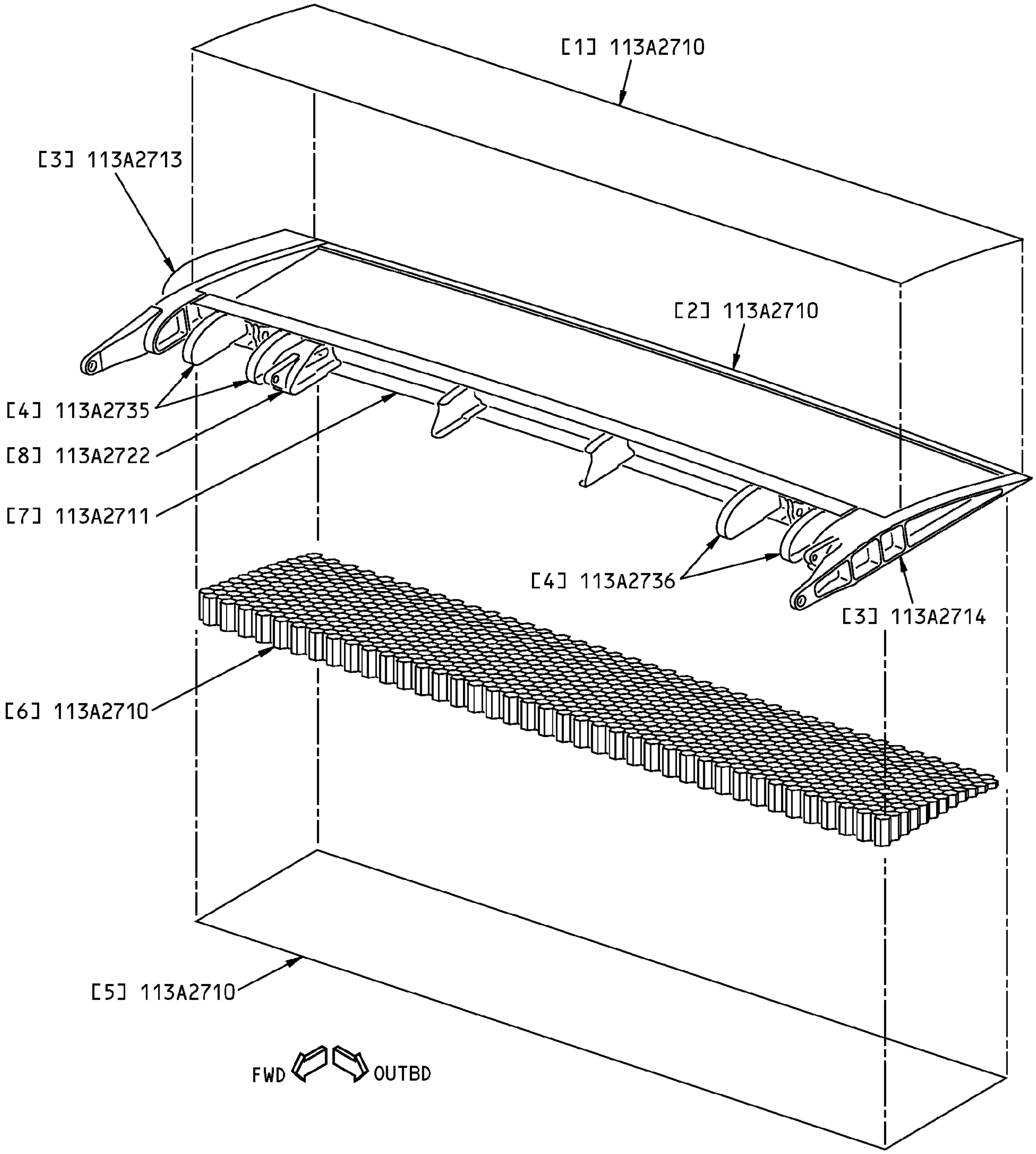

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Trailing Edge Movable Flap Locations**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A3001 | Flap Installation - Outboard Trailing Edge Flap |
| 113A3002 | Flap Assembly - Outboard Trailing Edge Flap |
| 113A2100 | Main Flap Final Assembly - Inboard Flap |
| 113A2700 | Aft Flap Assembly - Inboard Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A2710 | Bonded Assembly - Trailing Edge Wedge, Inboard Aft Flap |
| 113A2722 | Fitting Assembly - Inboard Pushrod, Inboard Aft Flap |
| 113A2725 | Fitting Assembly - Inboard Support Track, Inboard Aft Flap |
| 113A2726 | Fitting Assembly - Outboard Support Track, Inboard Aft Flap |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

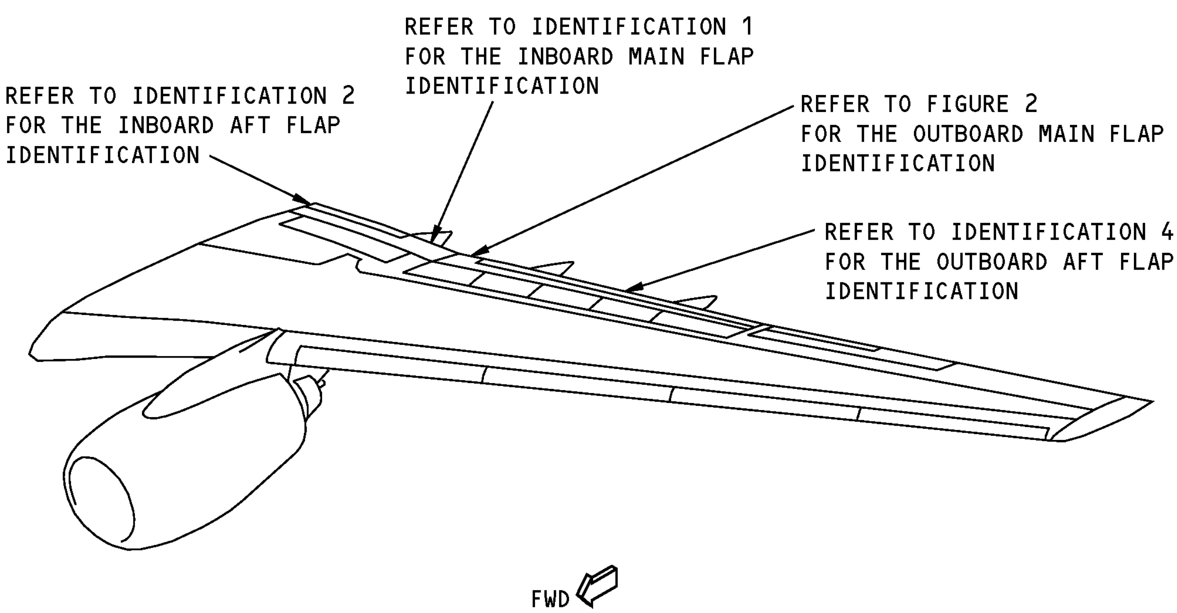
**Wing Trailing Edge Inboard Aft Flap Structure**
**Figure 2**

Table 2:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Skin - Upper | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| [2] | Trailing Edge Filler | | Epoxy impregnated glass woven fiber as given in BMS 8-79, Class III, Grade B, Style 1581 120, or 7781 | |
| [3] | Rib - Closure | | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Rib | 0.90 (22.86) | 7050-T7451 plate as given in AMS 4050 | |
| [5] | Skin - Lower | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |
| | | | Glass fabric overlay as given in BMS 8-79, class III, style 7781 or BMS 9-3, type H-2 as given in BAC 5337 | For cum line numbers 1131 and on |
| | | | (Optional) Glass fabric overlay as given in BMS 8-79, class III, style 7781 or BMS 9-3, type H-2 as given in BAC 5337 | For cum line numbers 1 through 1130 |
| [6] | Honeycomb Core | | Aluminum honeycomb core as given in BMS 4-4, Type 4-25, Class NPA, Form B, Grade I | |
| [7] | Spar | | 7075-T6 sheet as given in QQ-A-250/12 | |
| [8] | Fitting | | 7075-T6 sheet as given in QQ-A-250/12 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 3 - WING TRAILING EDGE OUTBOARD MAIN FLAP STRUCTURE



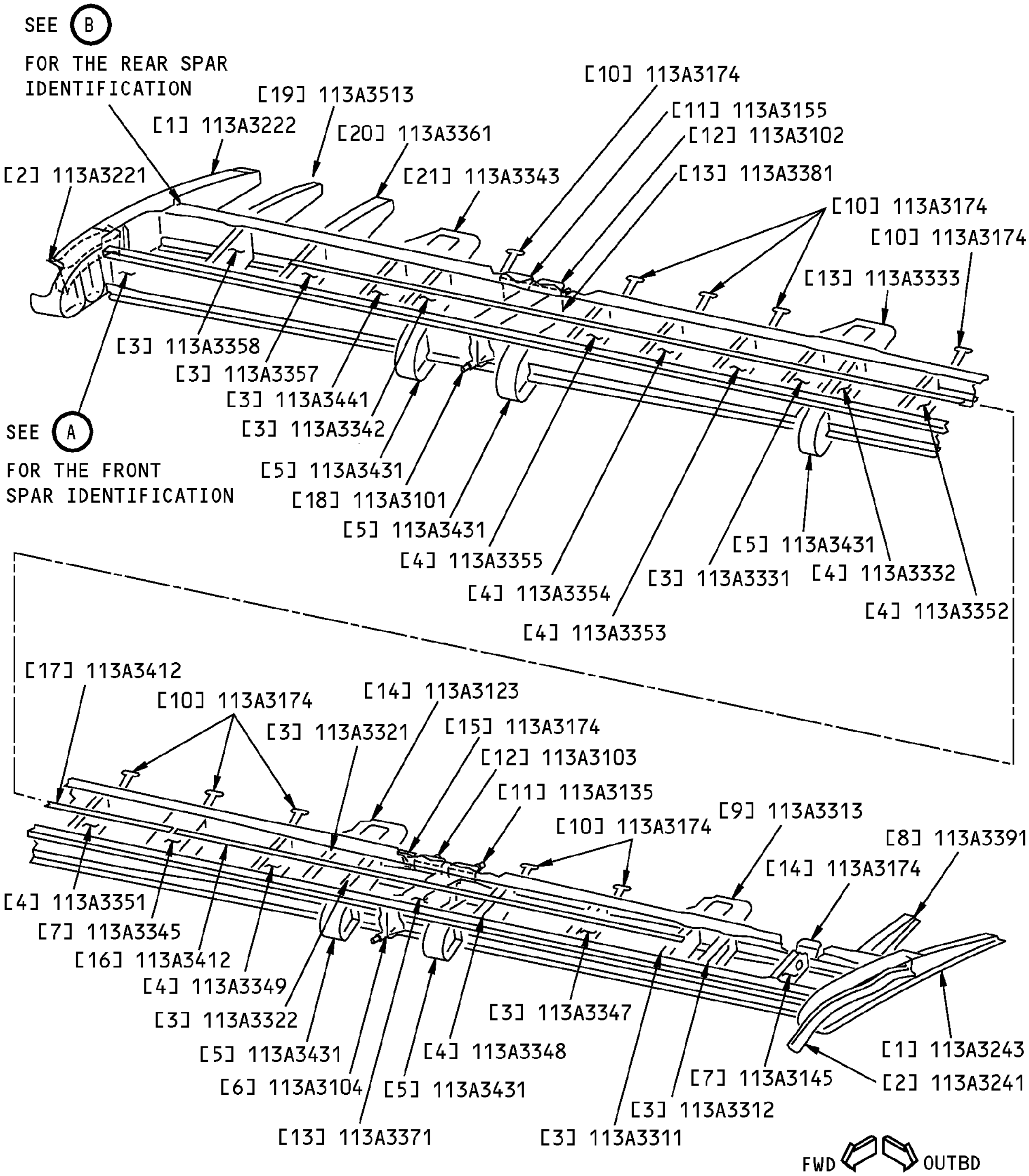

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Trailing Edge Movable Flap Locations**
**Figure 1**

**Table 1**:

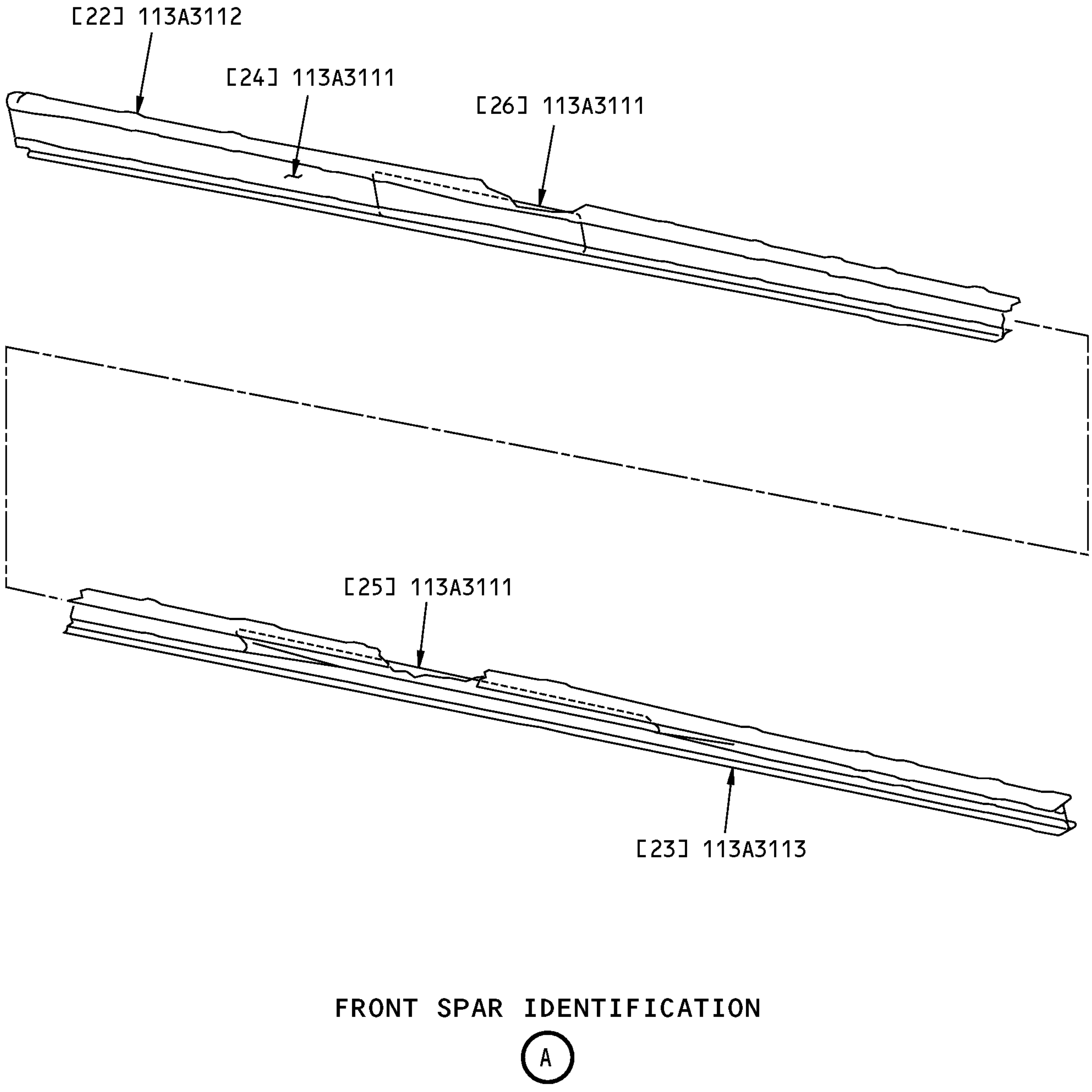
| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A3001 | Flap Installation Outboard Trailing Edge Flap |
| 113A3002 | Flap Assembly - Outboard Trailing Edge Flap |
| 113A3100 | Main Flap Assembly - Outboard Trailing Edge Flap |
| 113A3135 | Failsafe Fitting Assembly, Outboard Trailing Edge Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A3310 | Rib Assembly - Aft Track #1 Support Outboard Trailing Edge Flap |
| 113A3320 | Rib Assembly - Aft Track #2 Support Outboard Trailing Edge Flap |
| 113A3330 | Rib Assembly - Aft Track #3 Support Outboard Trailing Edge Flap |
| 113A3340 | Rib Assembly - Aft Track #4 Support Outboard Trailing Edge Flap |

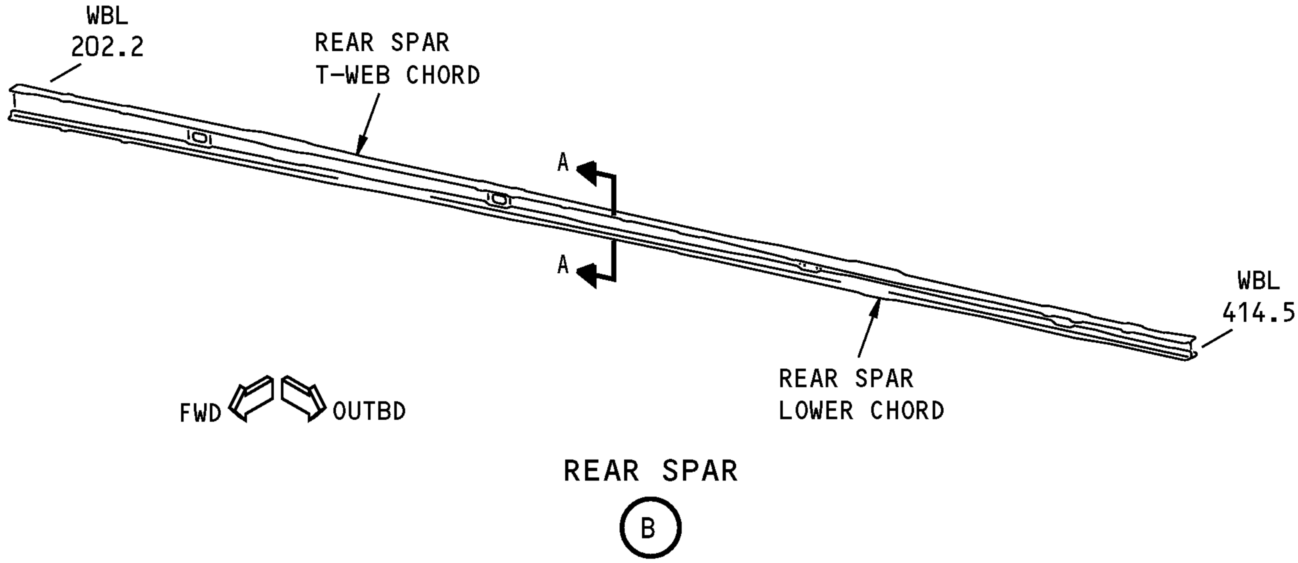


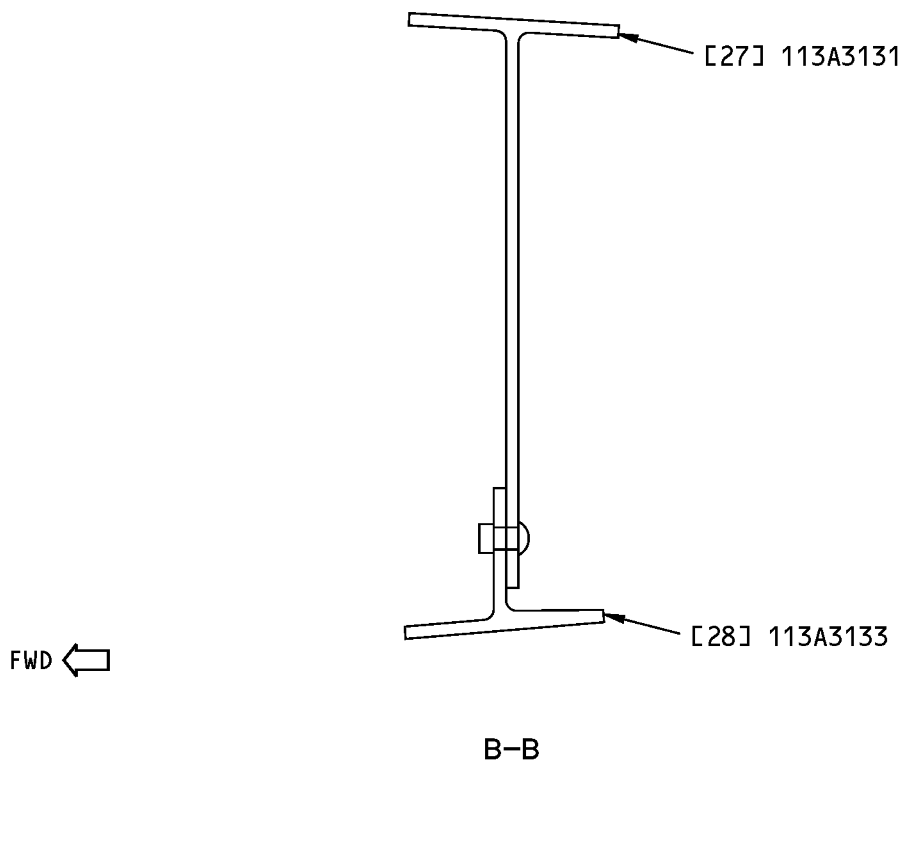

NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Wing Trailing Edge Outboard Main Flap Structure Identification**
**Figure 2 (Sheet 1 of 3)**




**Wing Trailing Edge Outboard Main Flap Structure Identification**
**Figure 2 (Sheet 2 of 3)**



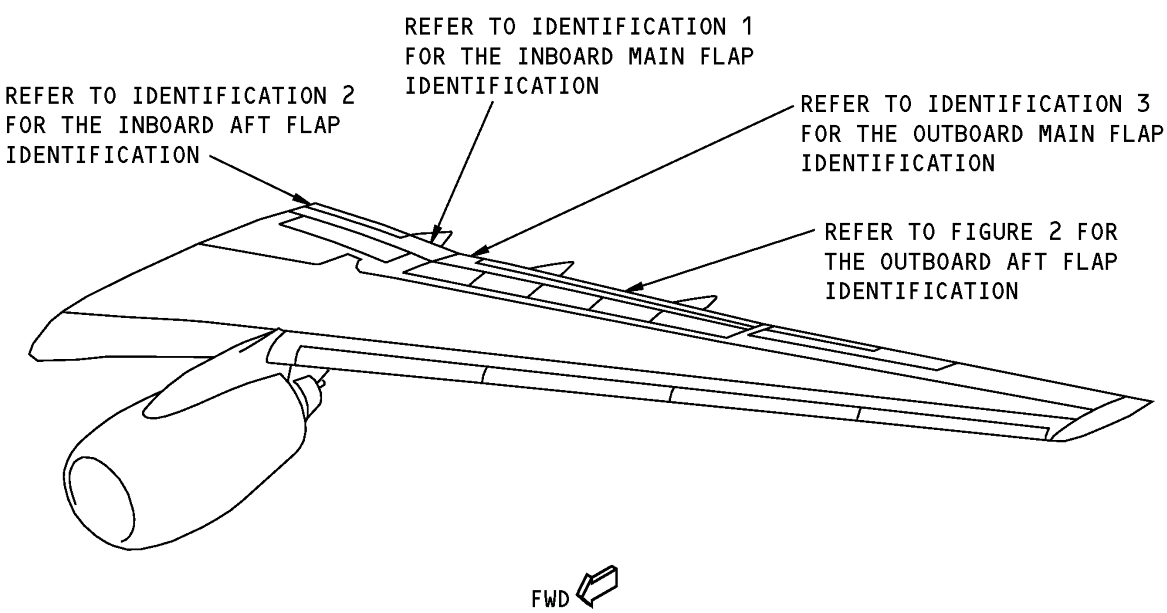

**Wing Trailing Edge Outboard Main Flap Structure Identification**
**Figure 2 (Sheet 3 of 3)**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Rib - Closure | | 7050-T7451 plate as given in AMS 4050 | |
| [2] | Rib - Deflection Control | | 15-5PH bar as given in AMS 5659, heat treat to 180-200 KSI | |
| [3] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [5] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [6] | Fitting | | 15-5PH bar as given in AMS 5659, heat treat to 180-200 KSI | |
| [7] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [8] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [9] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [10] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [11] | Fitting - Failsafe | | Ti-6Al-4V bar as given in Mil-T-9047 | |
| [12] | Fitting - Support | | 15-5PH bar as given in AMS 5659, heat treat to 180-200 KSI | |
| [13] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [14] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [15] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [16] | Stiffener | | BAC1503-100892 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [17] | Stiffener | | BAC1503-100413 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [18] | Fitting | | 15-5PH bar as given in AMS 5659, heat treat to 180-200 KSI | |
| [19] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [20] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [21] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [22] | Chord - Upper | | BAC1505-100692 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [23] | Chord - Lower | | BAC1505-100691 2024-T3511 extrusion as given in QQ-A-200/3 | |
| [24] | Web | 0.050 (1.27) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [25] | Doubler | 0.025 (0.64) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [26] | Doubler | 0.080 (2.03) | 2024-T3 sheet as given in QQ-A-250/4 | |
| [27] | Chord - Upper Tee | | BAC1505-100690 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [28] | Chord - Lower | | BAC1505-100689 2024-T3511 extrusion as given in QQ-A-200/3 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 4 - WING TRAILING EDGE OUTBOARD AFT FLAP STRUCTURE



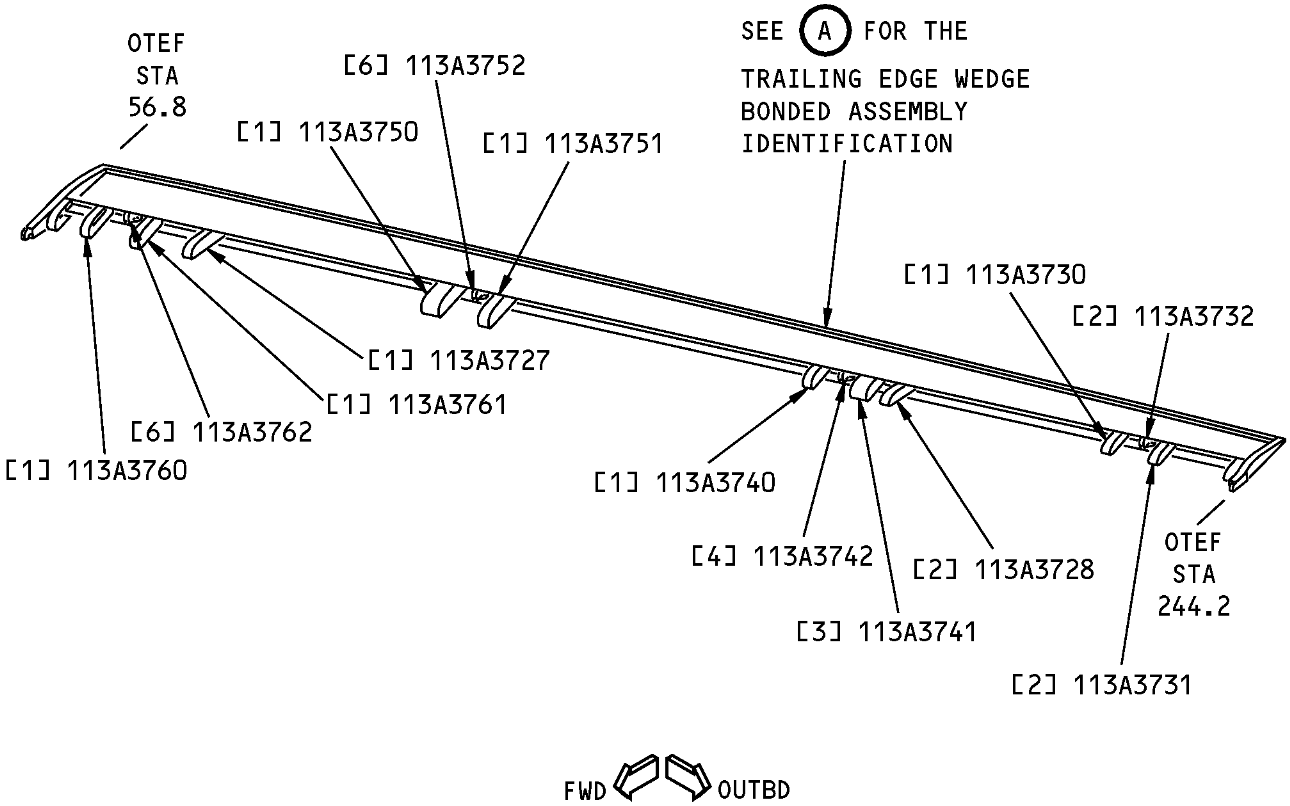

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Wing Trailing Edge Movable Flap Locations**
**Figure 1**

**Table 1**:

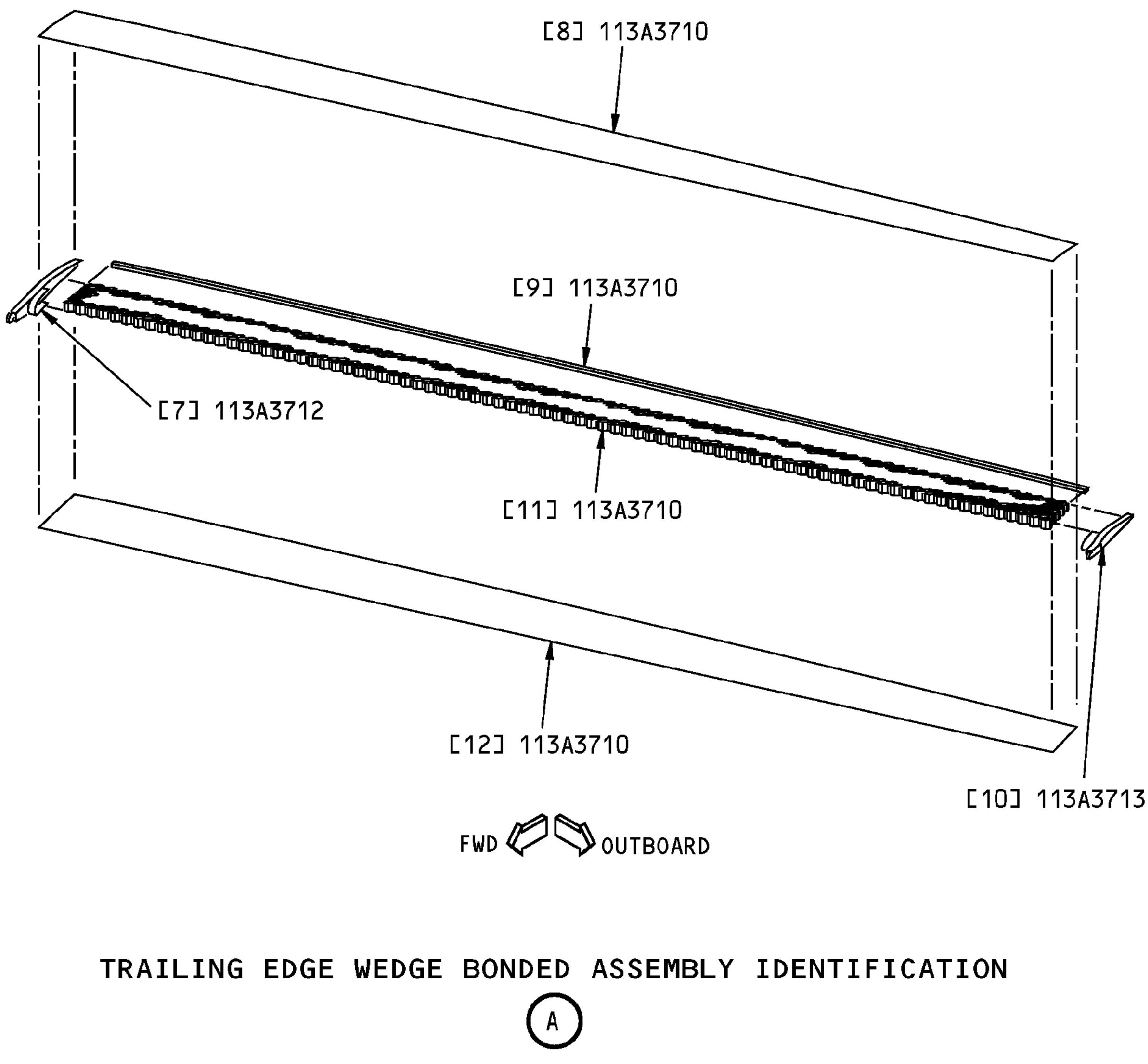
| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable T.E. Functional Product Collector |
| 113A3001 | Flap Installation - Outboard Trailing Edge Flap |
| 113A3002 | Flap Assembly - Outboard Trailing Edge Flap |
| 113A3700 | Aft Flap Assembly - Outboard Trailing Edge Flap |
| 113A3725 | Drive Fitting Assembly - Inboard, Outboard Aft Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A3726 | Drive Fitting Assembly - Outboard, Outboard Aft Flap |
| 113A3732 | Fitting Assembly - Track #1 Support, Outboard Aft Flap |
| 113A3742 | Fitting Assembly - Track #2 Support, Outboard Aft Track |
| 113A3752 | Fitting Assembly - Track #3 Support, Outboard Aft Flap |
| 113A3762 | Fitting Assembly - Track #4 Support, Outboard Aft Flap |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Wing Trailing Edge Outboard Aft Flap Structure**
**Figure 2 (Sheet 1 of 2)**




Wing Trailing Edge Outboard Aft Flap Structure
Figure 2 (Sheet 2 of 2)

**Table 2**:

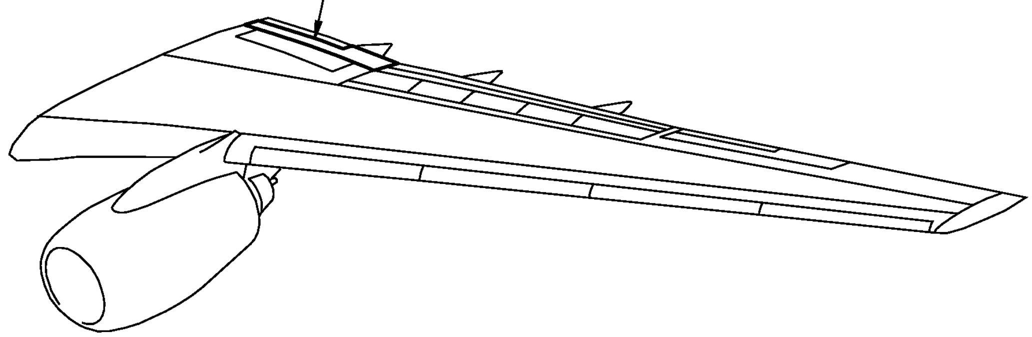
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [2] | Fitting | | 7050-T7451 plate as given in AMS 4050 | |
| [3] | Rib | | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Fitting | | 7050-T7451 plate as given in AMS 4050 | |
| [5] | Fitting | | 7050-T7451 plate as given in AMS 4050 | |
| [6] | Fitting | | 7050-T7451 plate as given in AMS 4050 | |
| [7] | Rib - Closure | | 7050-T7451 plate as given in AMS 4050 | |
| [8] | Skin - Upper | 0.016 (0.41) | 7075-T6 sheet as given in QQ-A-250/12 | |
| [9] | Fillet | | Glass Fiber Reinforced Plastic (GFRP) Pre-preg as given in BMS 8-79, Style 1581 | |
| [10] | Rib - Closure | | 7050-T7451 plate as given in AMS 4050 | |
| [11] | Honeycomb Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 4-25 NPA | |
| [12] | Skin - Lower | 0.020 (0.51) | 7075-T6 sheet as given in QQ-A-250/12 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

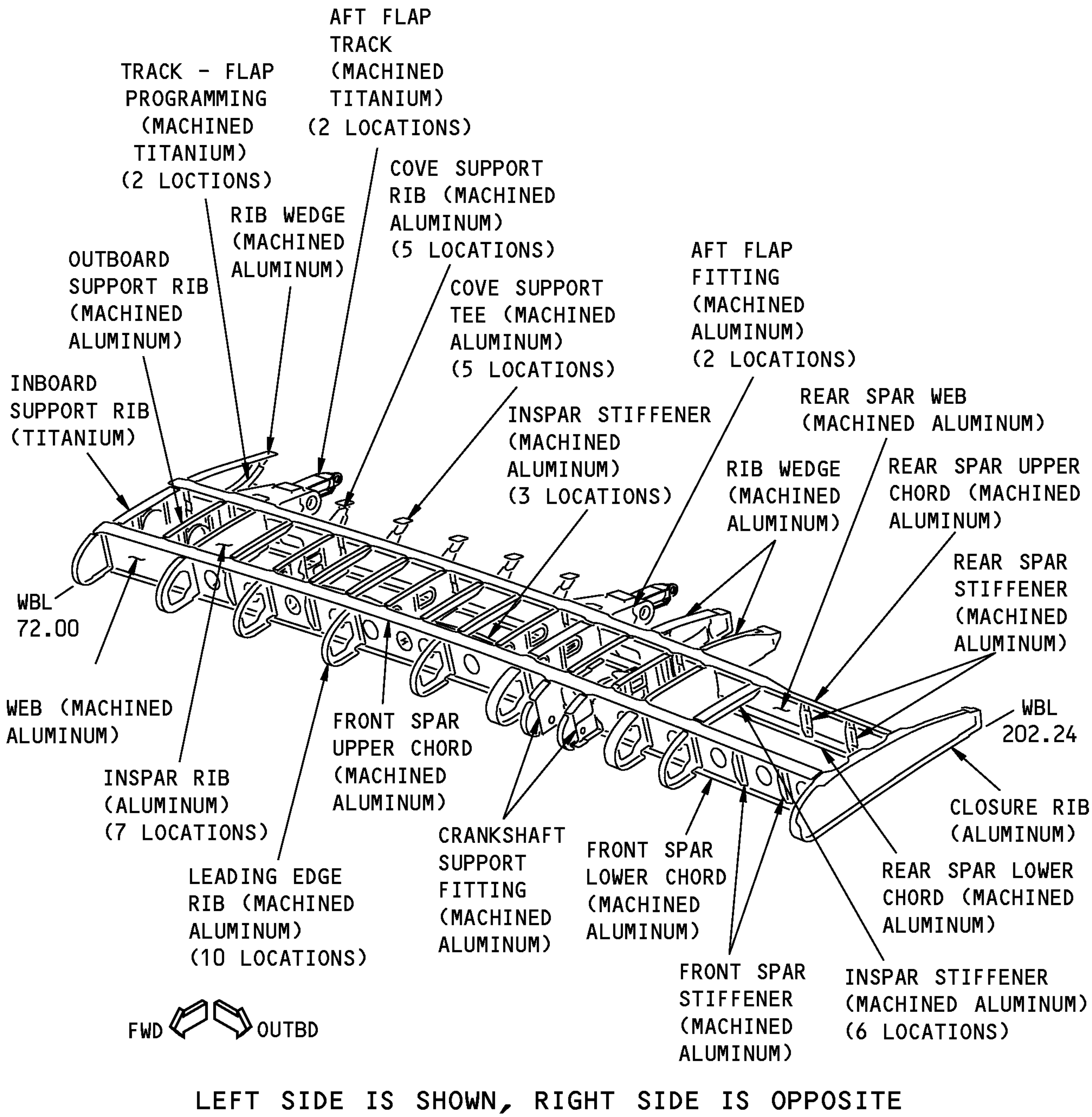
## ALLOWABLE DAMAGE 1 - WING INBOARD TRAILING EDGE MAIN FLAP STRUCTURE

1. **Applicability**

   A. This subject gives the allowable damage limits for the Main Flap Structure of the Wing Inboard Trailing Edge shown in Wing Inboard Trailing Edge Main Flap Structure Location, Figure 101/ALLOWABLE DAMAGE 1.




**Wing Inboard Trailing Edge Main Flap Structure Location**
**Figure 101**




G04443 S0006594934_V2

**Wing Inboard Trailing Edge Main Flap Structure**
**Figure 102**

## 2. General

**WARNING:** SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON, OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STEAM EXPLOSION. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

A. Do the steps that follow if you have damage to the aluminum or titanium parts:

(1) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do as follows:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the rear spar chords, support ribs, aft flap fittings, aft flap tracks, and closure ribs.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum parts. Refer to 51-20-01.

(3) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area of aluminum parts. Refer to SOPM 20-41-02.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-10-02, GENERAL | Inspection and Removal of Damage |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-10-03 | General - Shot Peening Procedures |

(Continued)

| Reference | Title |
|---|---|
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

**4. Allowable Damage Limits**

A. Front Spar Web and Rear Spar Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , E , G , and H .

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail I .

**NOTE**: Fill the space between the flange and skin with BMS 5-95 sealant if the space is more than 0.005 inch.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch maximum in diameter

2) A minimum of 2.0 inches away from a fastener hole, the edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

a) Install the rivet without sealant.

B. Front and Rear Spar Chords, Ribs, Rib Wedge, Stiffeners, Cove Support Tees, and the Flap Programming Track.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , F , G , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Crank Shaft Support Fitting and Aft Flap Fitting

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , F , G , H , and J .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Aft Flap Track

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details K, and L.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

(5) Wear:

(a) Horizontal flange thickness wear:

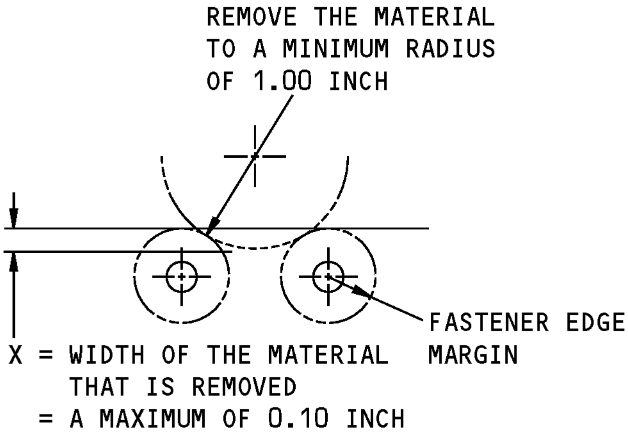
1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail L

a) This is an interim repair if the tungsten carbide coating is worn away and the base metal is exposed. Inspect the repair as given in Paragraph 5./ALLOWABLE DAMAGE 1

b) This is a permanent repair if the tungsten carbide coating is not worn away.

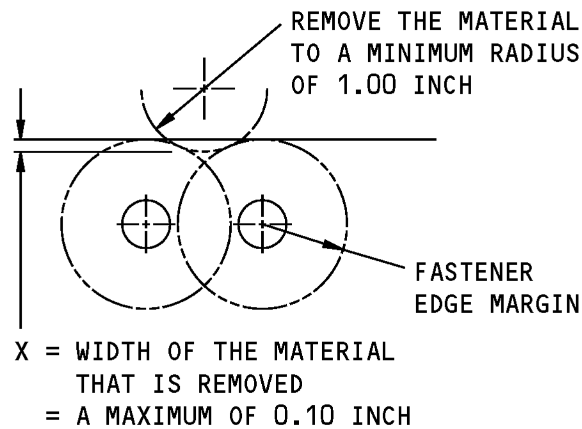
(b) Horizontal flange edge wear:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail K



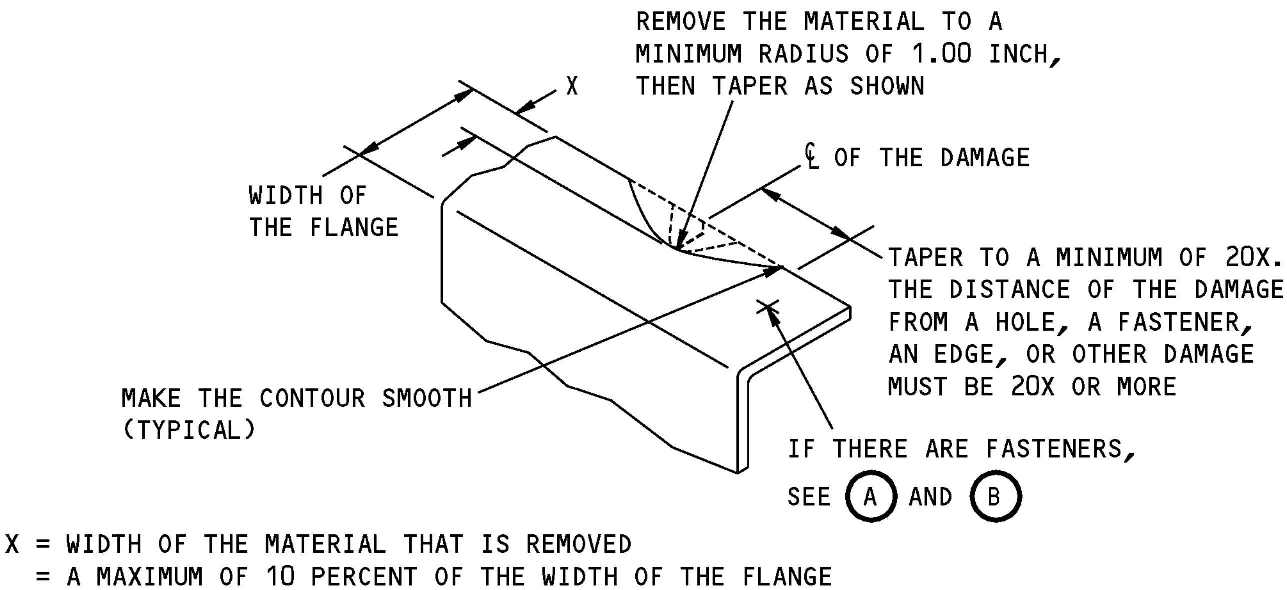

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

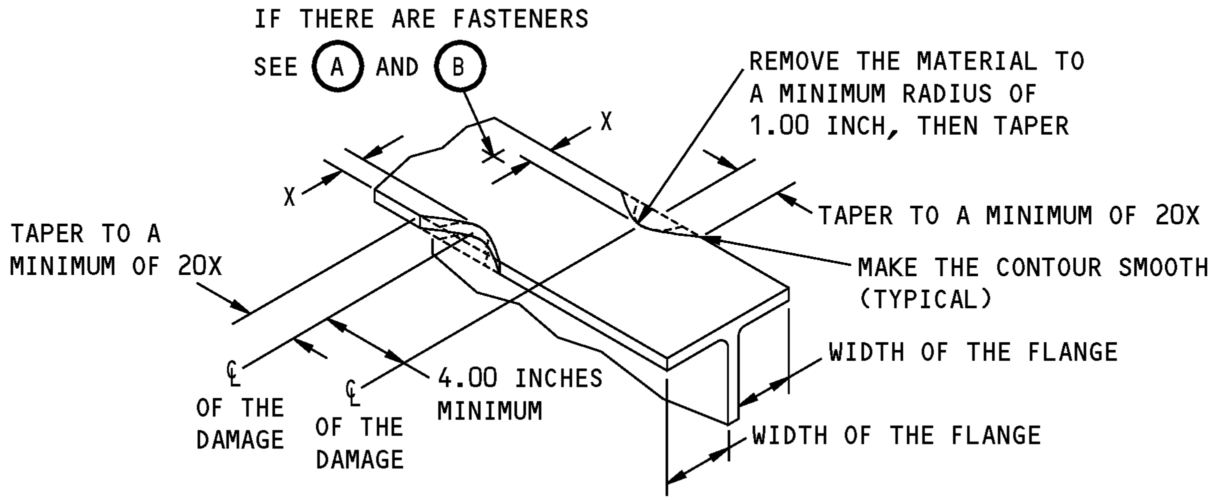
B




REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

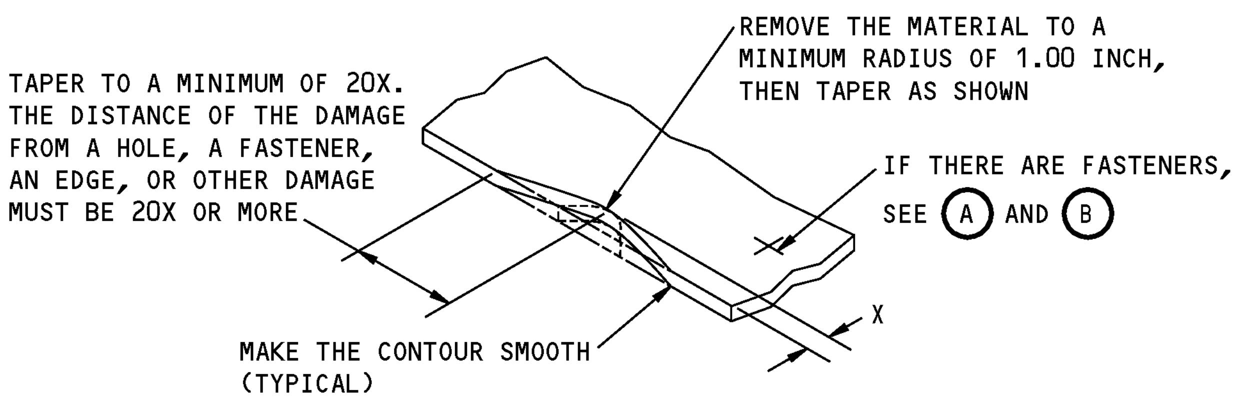
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 6)**




X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

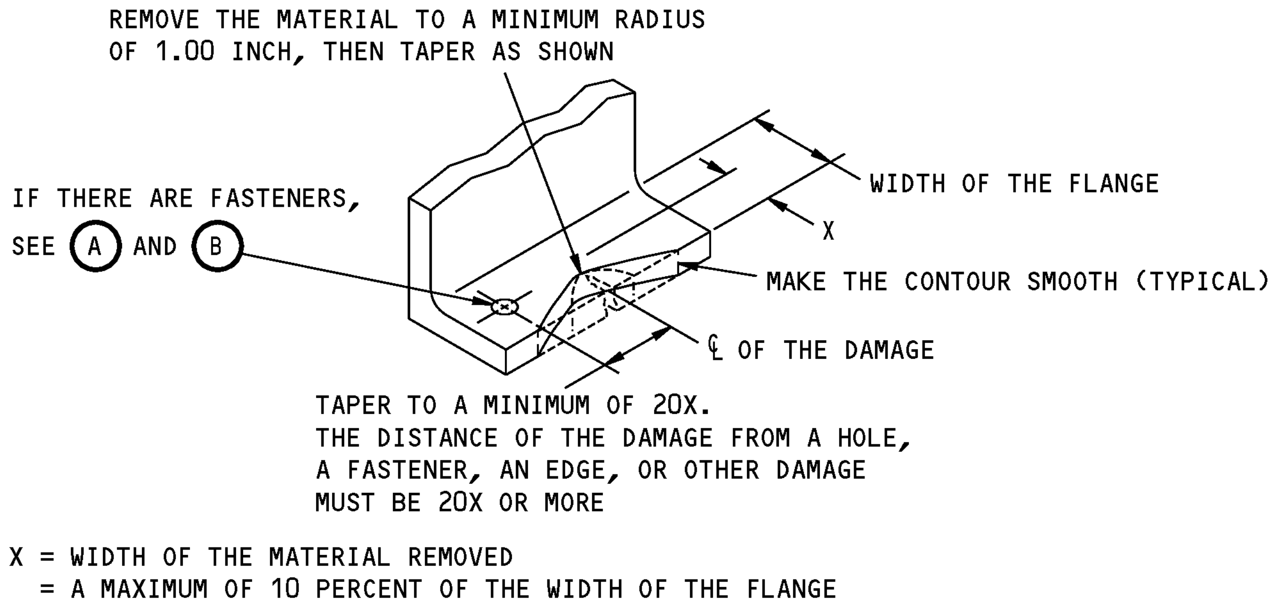



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

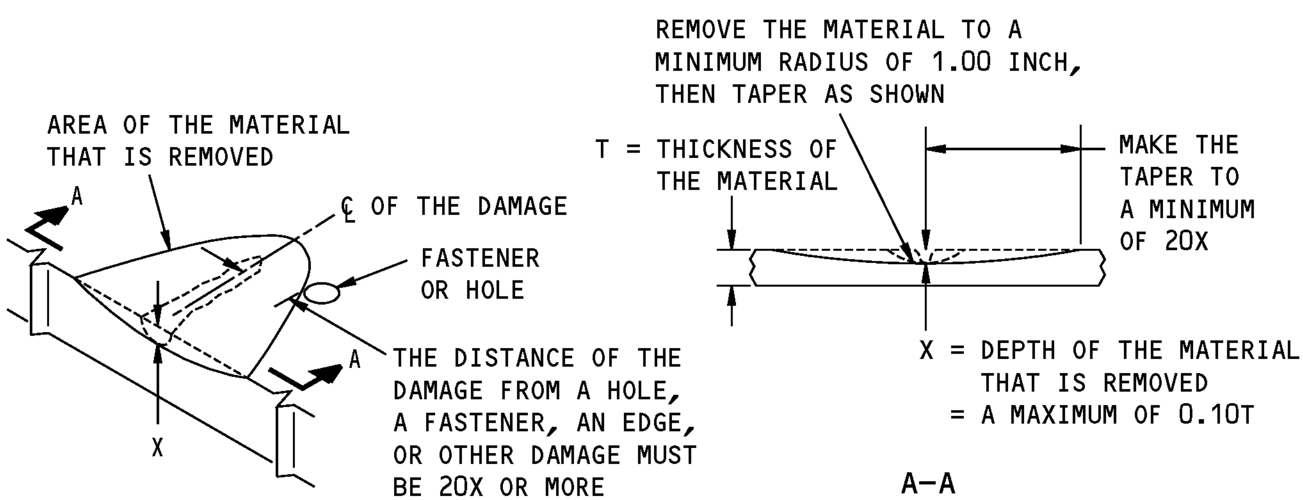
E

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 6)**




REMOVAL OF DAMAGED MATERIAL AT AN EDGE

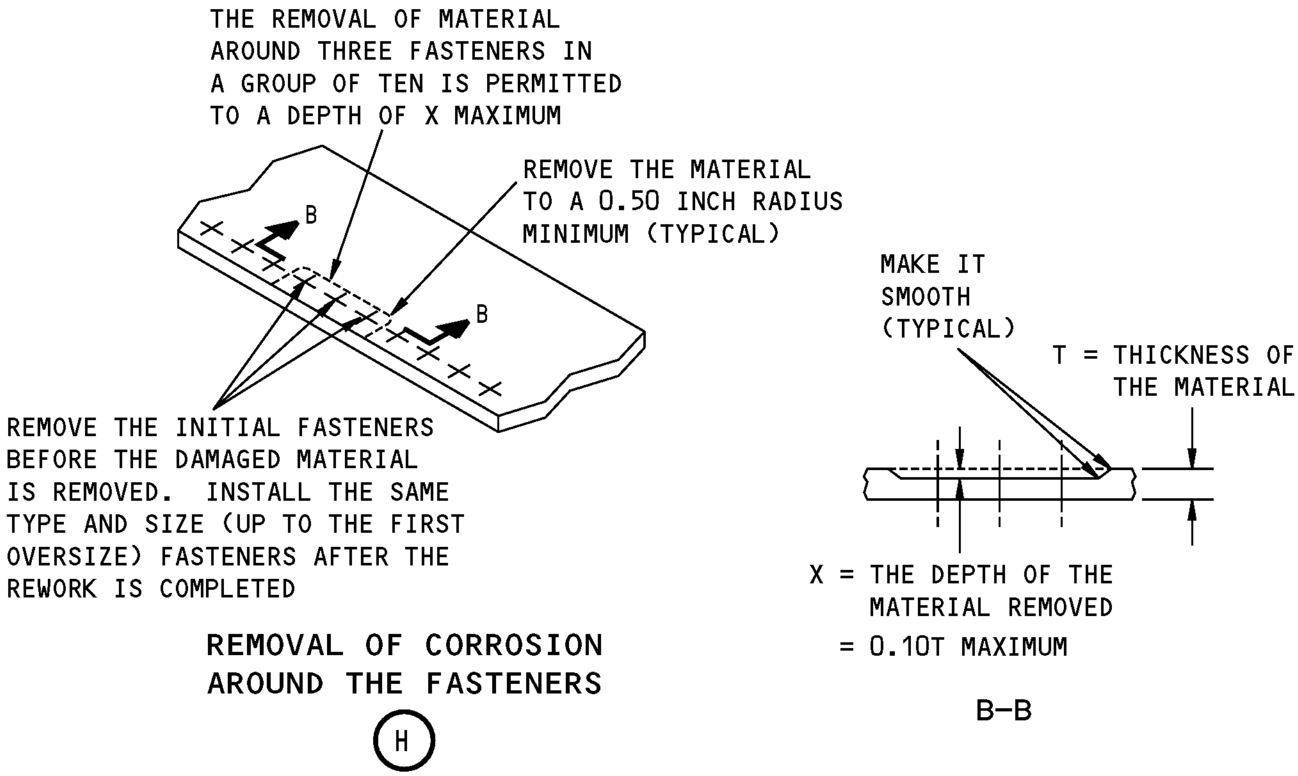
F

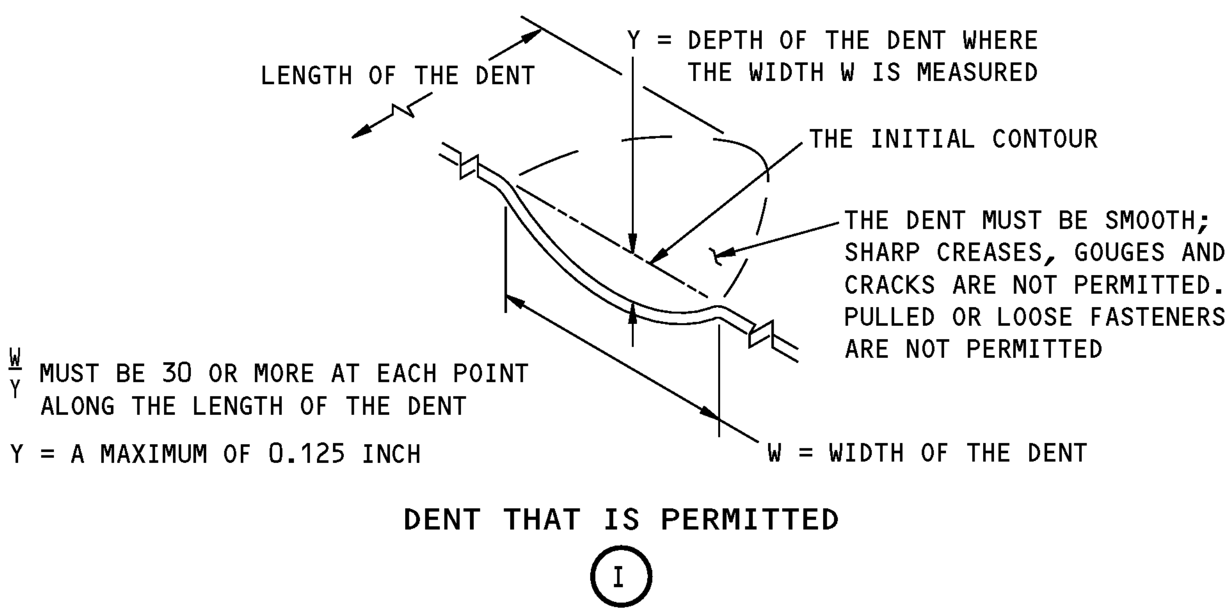



REMOVAL OF DAMAGED MATERIAL ON A SURFACE

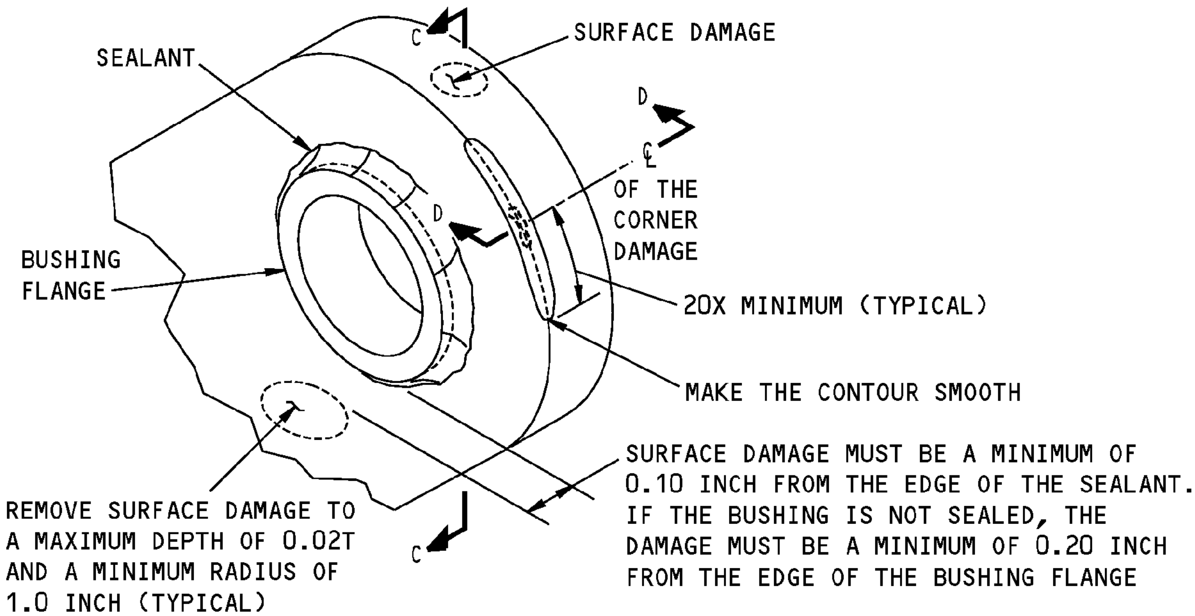
G

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 6)**



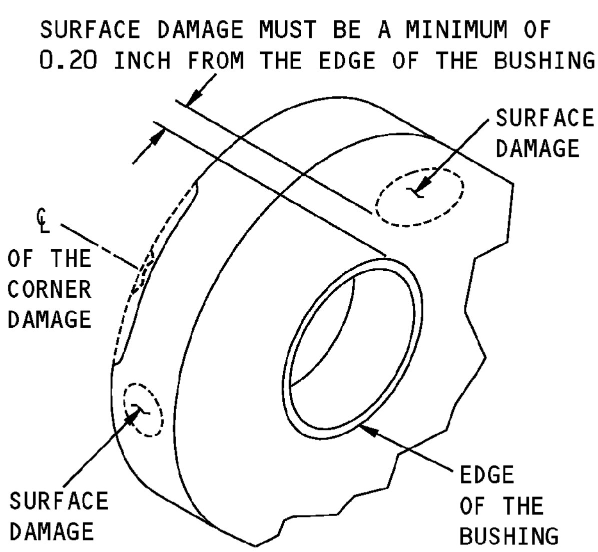

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 6)**



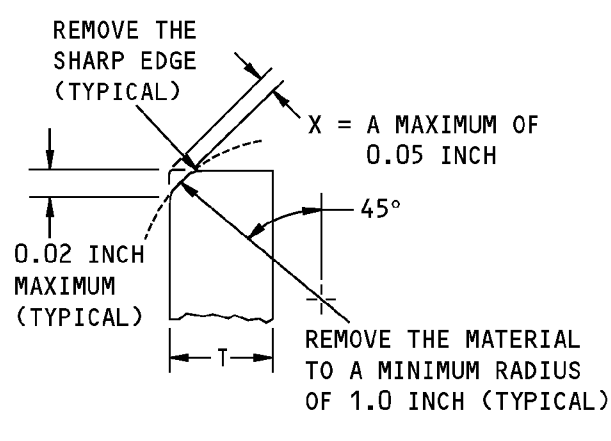

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

J

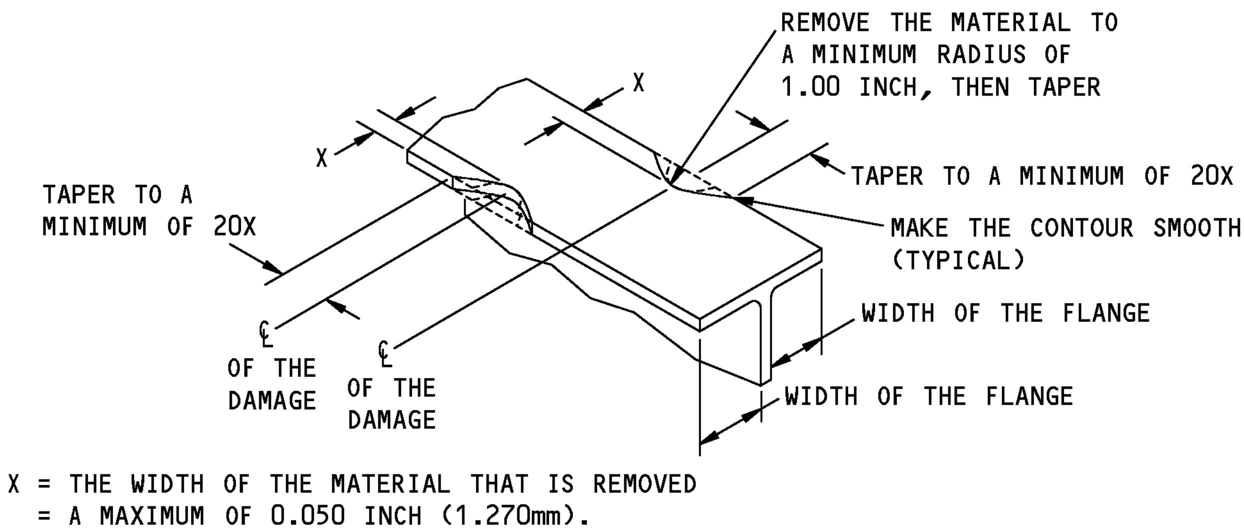



C-C



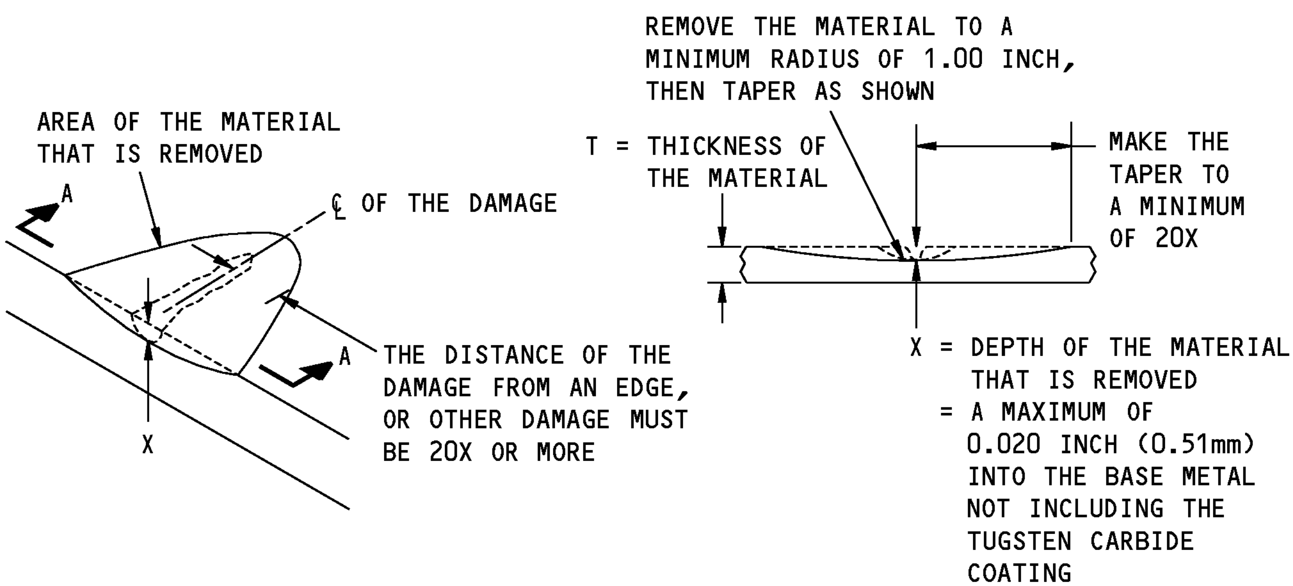

D-D

**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 6)**




REMOVAL OF DAMAGED MATERIAL ON AN EDGE

K




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

L

A-A

1543831 S0000282147_V1

**Allowable Damage Limits**
**Figure 103 (Sheet 6 of 6)**

5. **Inspection Requirements**

A. Do the inspection of the interim repair as given in Table 101/ALLOWABLE DAMAGE 1

**Table 101**: Interim Repair Inspection Requirements

| INSPECTION THRESHOLD | REPEAT INSPECTIONS | |
|---|---|---|
| | METHOD | INTERVAL |
| From the start of the repair | As given in 51-10-02, GENERAL | 6 Months or 1000 Flight Cycles whichever comes sooner |
| NOTES: Do a detailed inspection of the Aft Flap Track to ensure the horizontal flange thickness does not wear beyond a maximum of 0.020 in. (0.508 mm) into the base metal. The 0.020 in. (0.508 mm) limit applies to the base metal thickness reduction and does not include the tungsten carbide coating. | | |

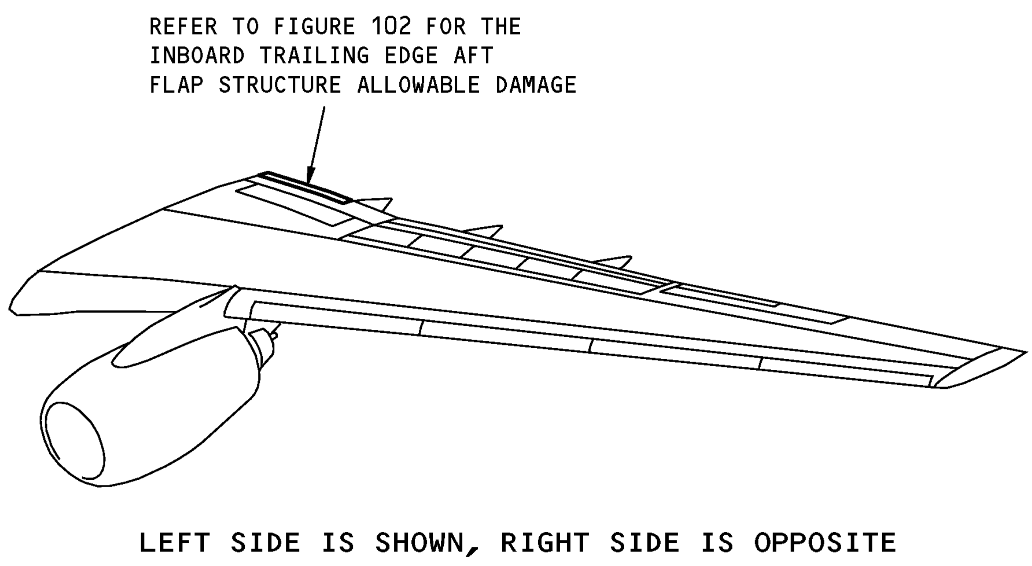
## ALLOWABLE DAMAGE 2 - WING INBOARD TRAILING EDGE AFT FLAP STRUCTURE

1. **Applicability**
   A. This subject gives the allowable damage limits for the wing inboard trailing edge aft flap structure shown in Wing Inboard Trailing Edge Aft Flap Location, Figure 101/ALLOWABLE DAMAGE 2.

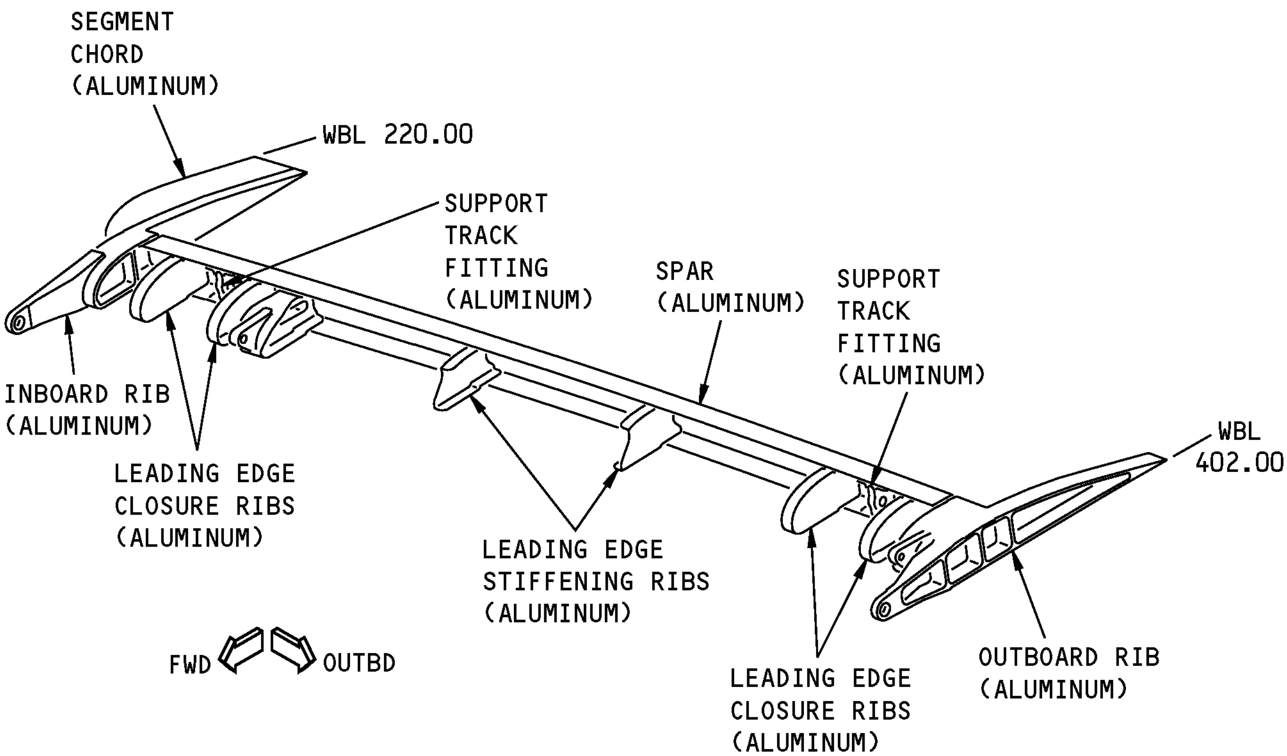
2. **General**
   A. Do the steps that follow if you have damage to the parts:
      (1) Remove the damage. Refer to 51-10-02 for the investigation and cleanup procedures.
      (2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.
      (3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.
   B. After you remove the damage, do as follows:

      **WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

      (1) Do not flap peen or shot peen the spar, trailing edge wedge, or the closure ribs.
         (a) Refer to 51-20-06 for the shot peen intensity and shot number.
         (b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.
      (2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.
      (3) Apply one layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-44-04.




**Wing Inboard Trailing Edge Aft Flap Location**
**Figure 101**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

INBOARD TRAILING EDGE AFT FLAP

**Wing Inboard Trailing Edge - Aft Flap Structure**
**Figure 102**

**3. References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| 57-53-01 | TRAILING EDGE FLAP SKIN |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

**4. Allowable Damage Limits**

A. Closure Ribs, Stiffener Ribs, Segment Chord, and Spar Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F , G , and H .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail I .

(a) If there is a gap between the flap skin and the structure, that is more than 0.005 inch, then fill the gap with a shim.

1) Refer to 57-53-01 for the dent limits for the flap skin.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

a) Install the rivet without sealant.

B. Inboard Rib, Outboard Rib, Spar Chords, and Fittings

(1) Cracks:

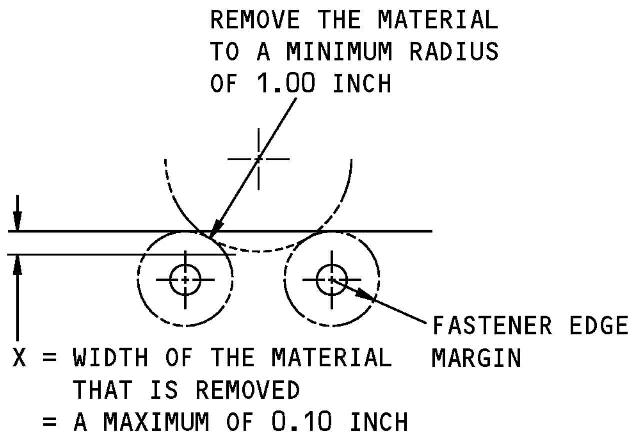
(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E.

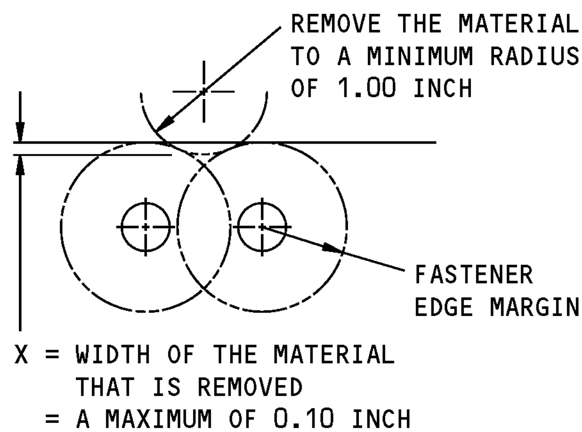
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



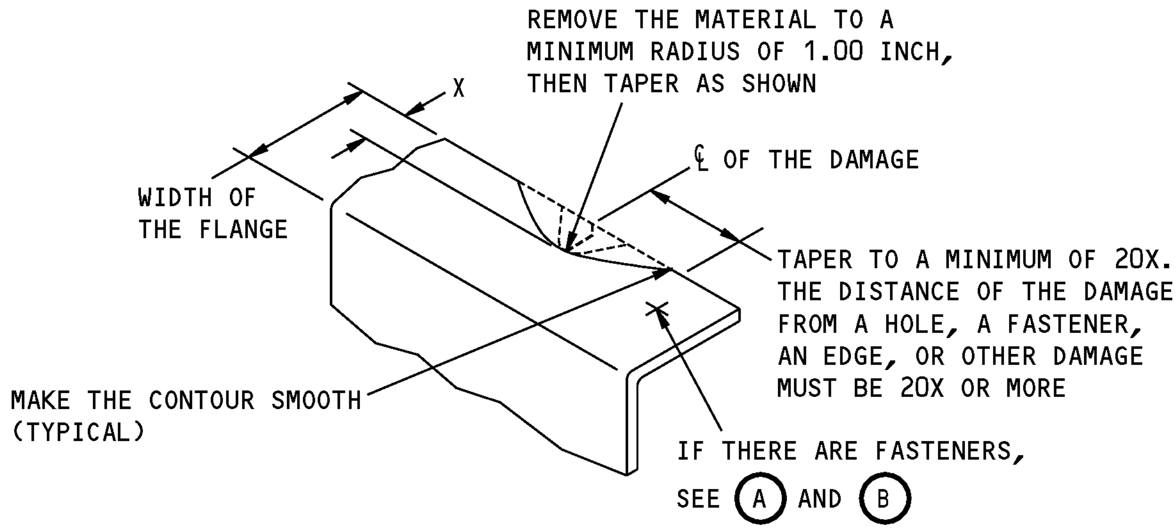

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

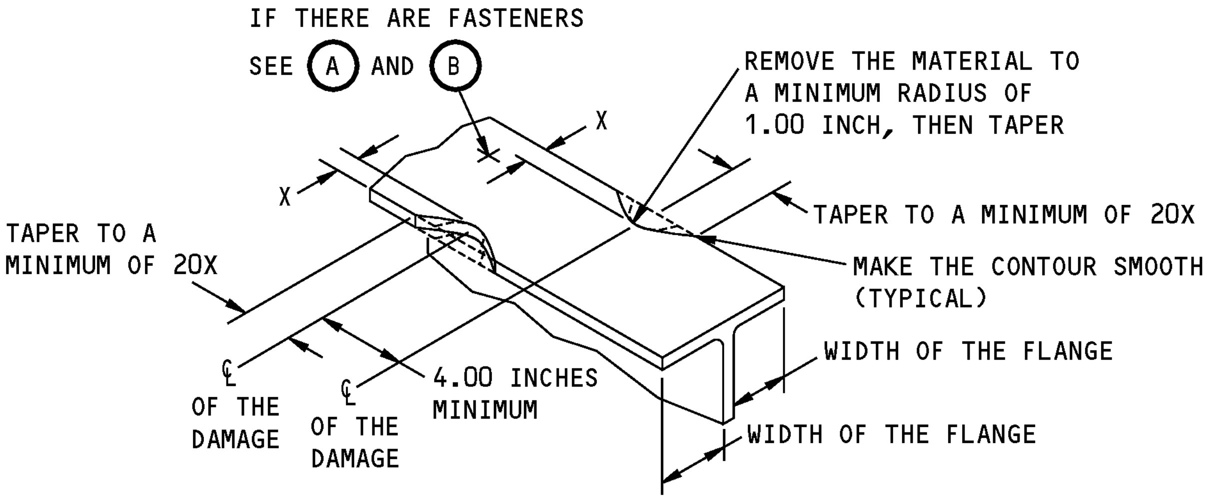



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

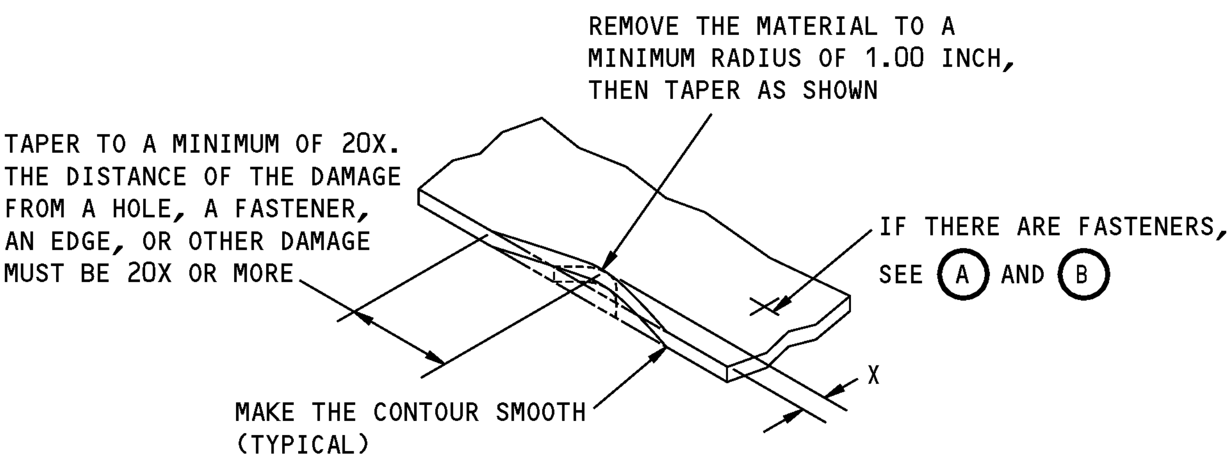
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 5)**




X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

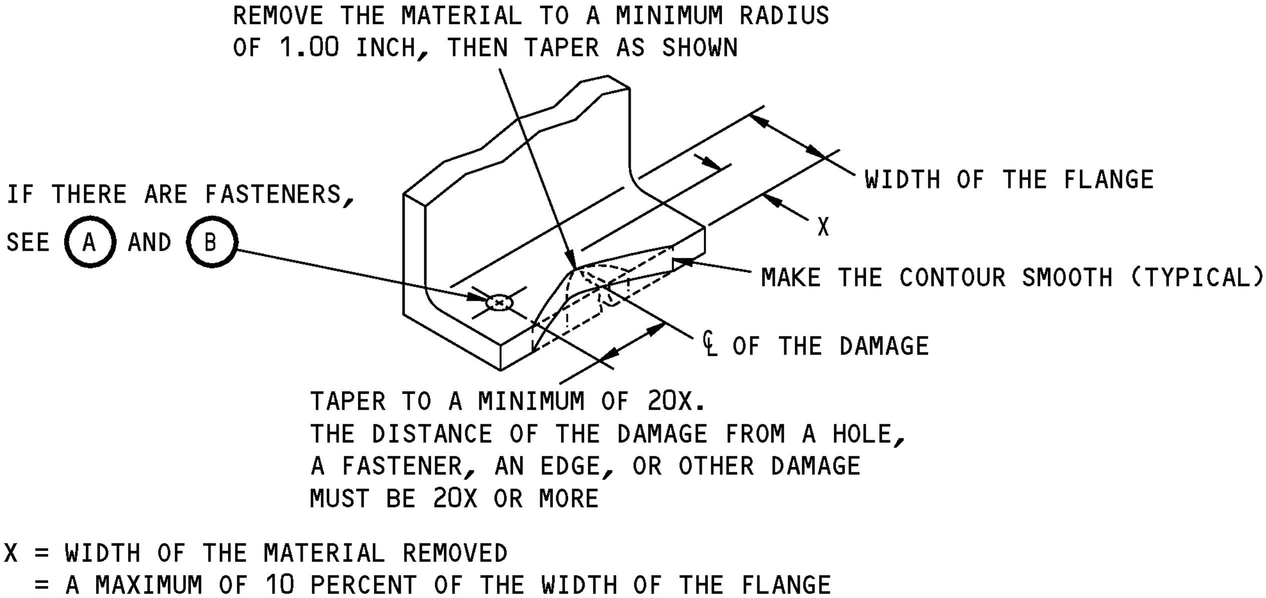



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

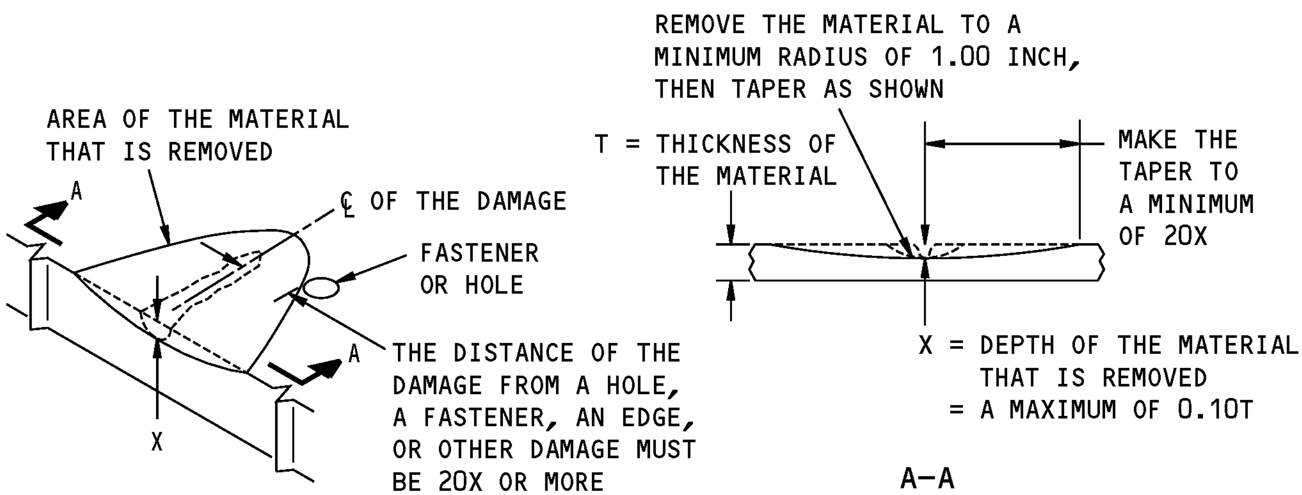
E

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 5)**




REMOVAL OF DAMAGED MATERIAL AT AN EDGE

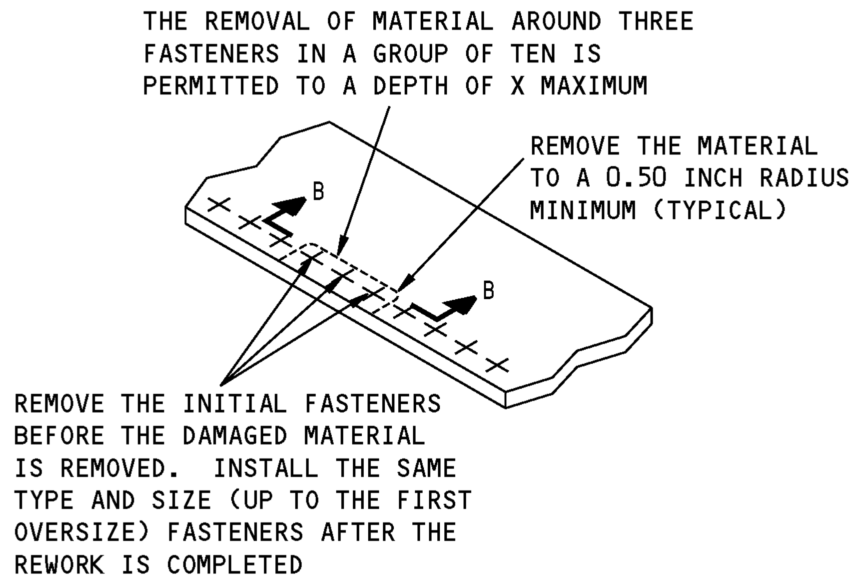
F




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

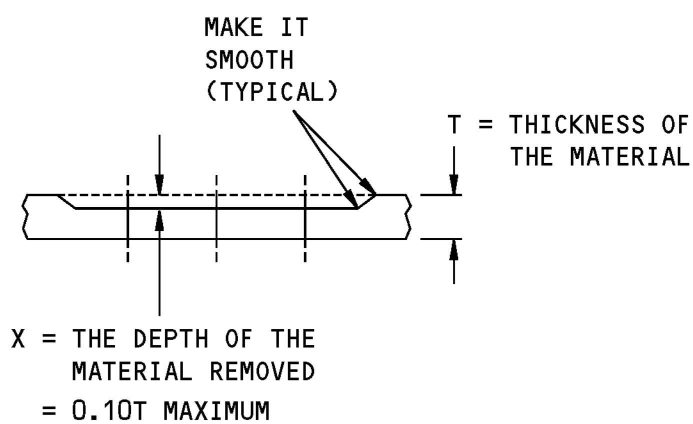
G

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 5)**



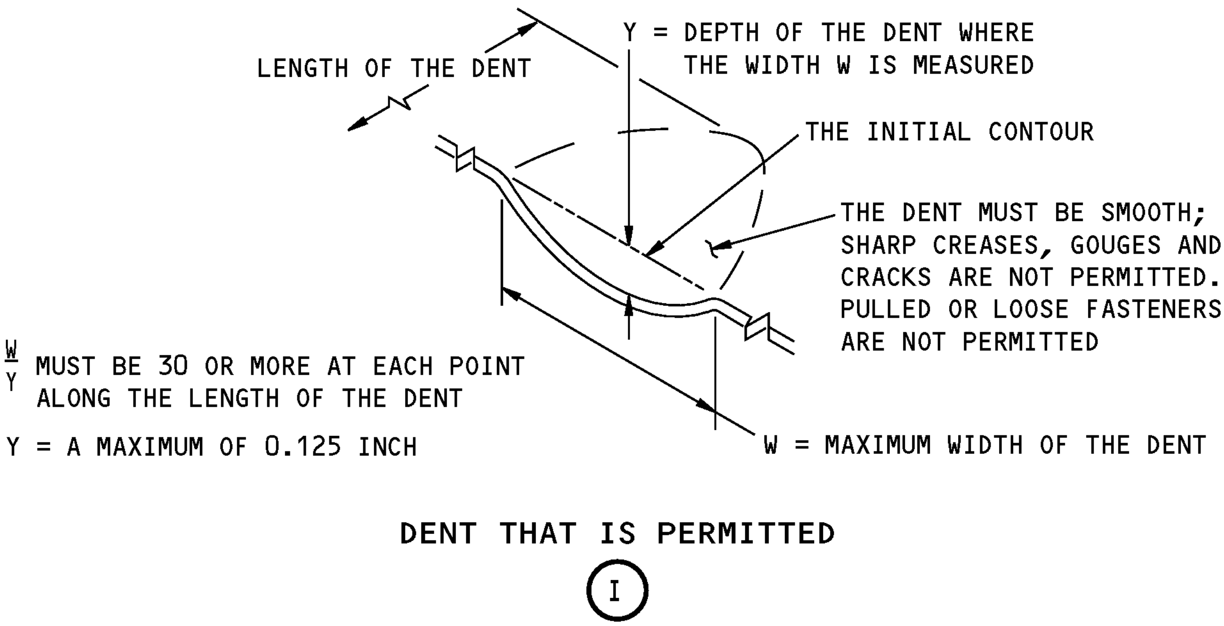

REMOVAL OF CORROSION AROUND THE FASTENERS

H

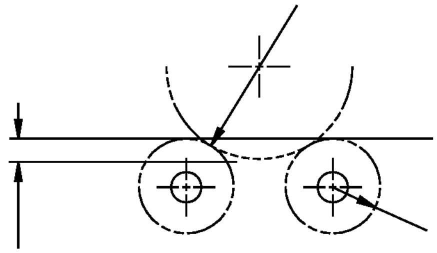



B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 5)**

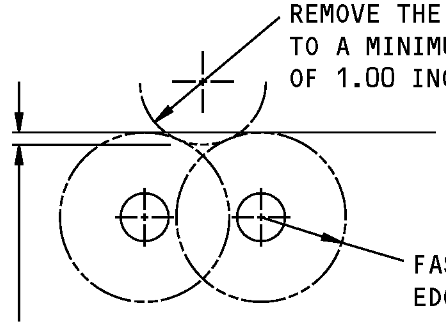



**Allowable Damage Limits**
**Figure 103 (Sheet 5 of 5)**




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

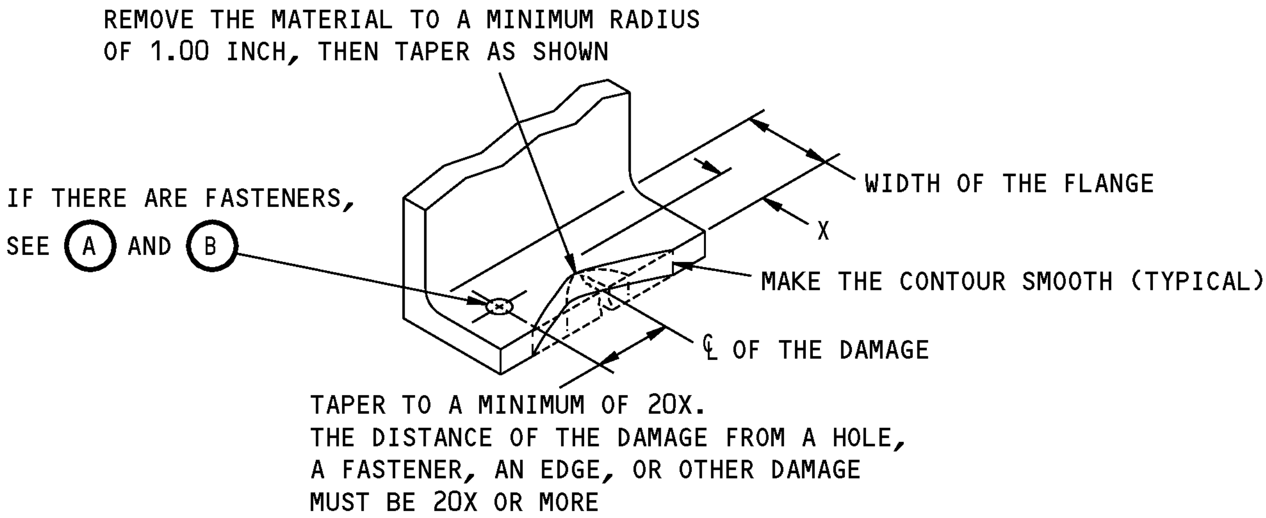
A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

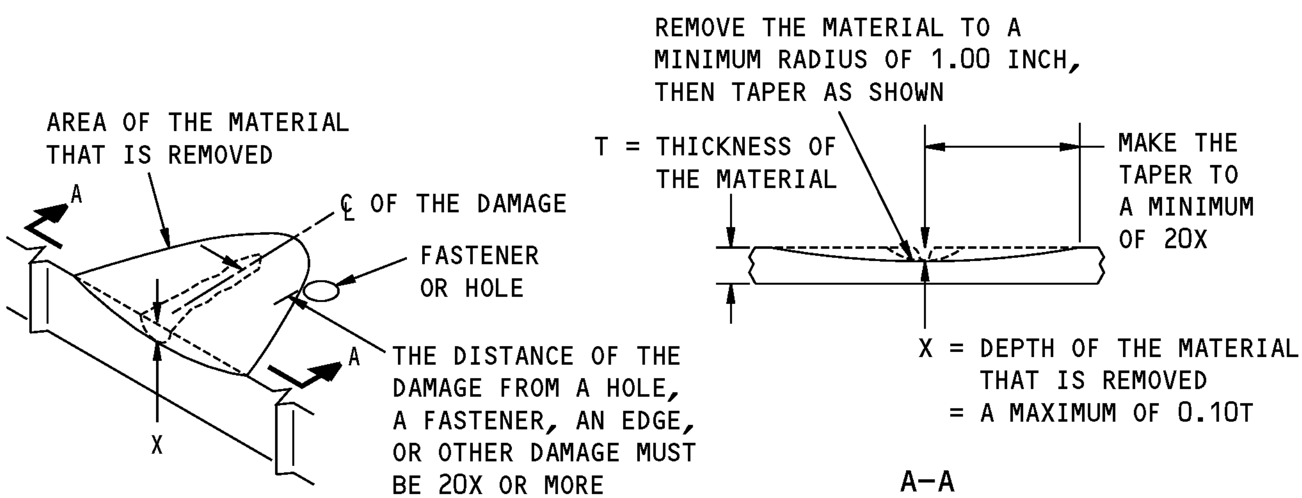
**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**




X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

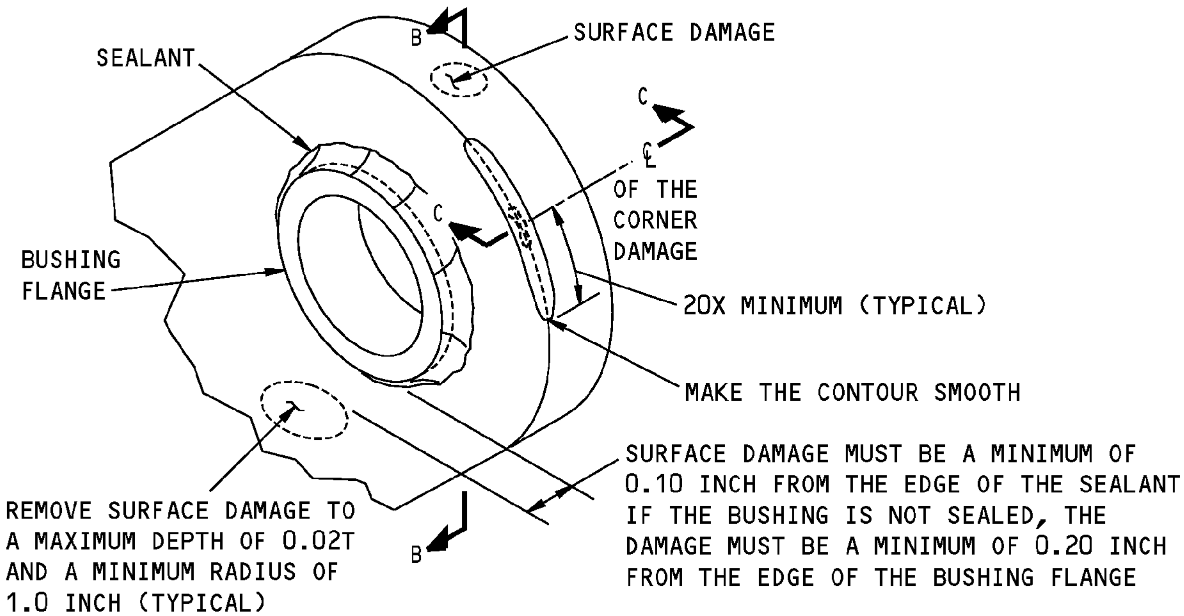
C




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

D

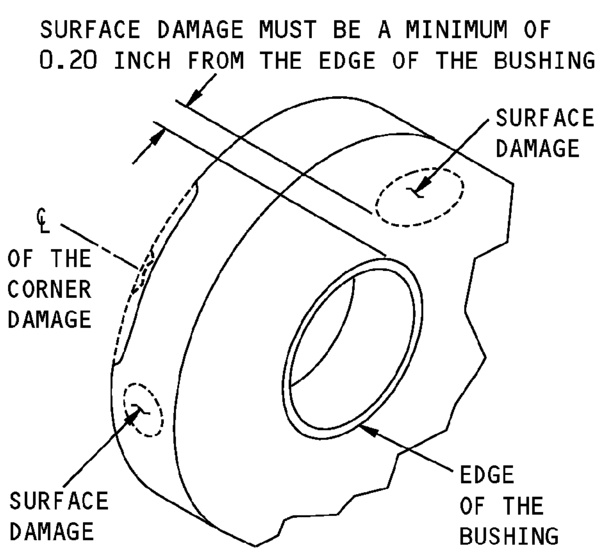
**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**



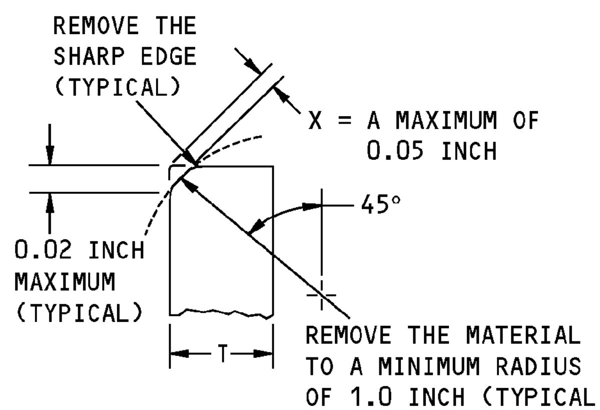

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

E




B-B



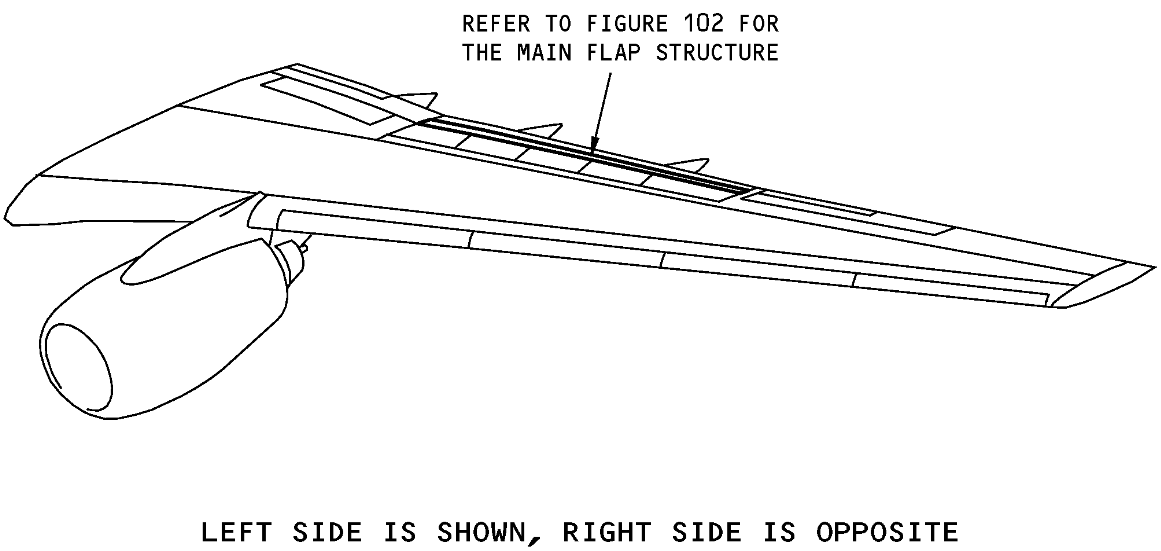

C-C

**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**

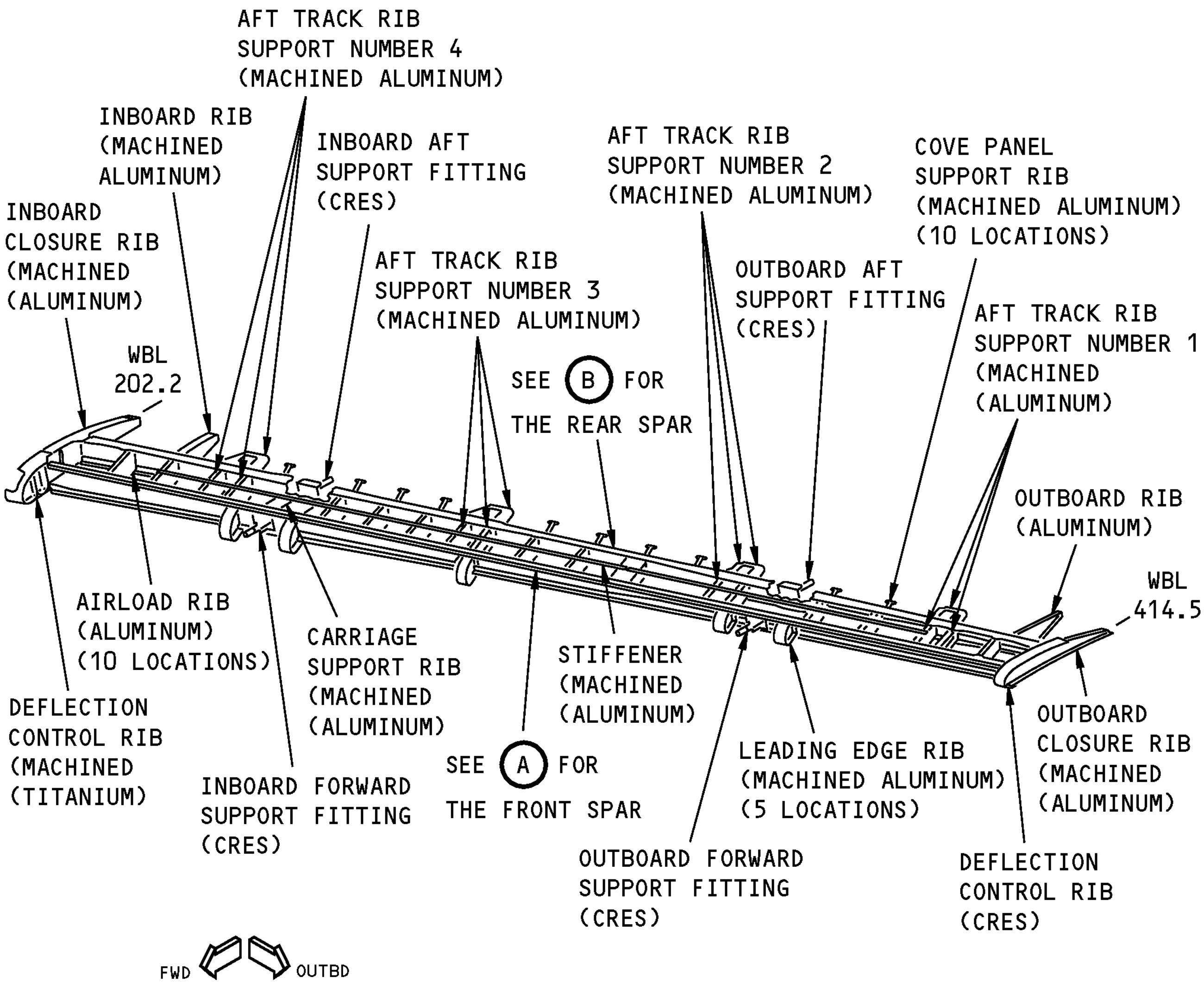
## ALLOWABLE DAMAGE 3 - WING OUTBOARD TRAILING EDGE MAIN FLAP STRUCTURE

### 1. Applicability

A. This subject gives the allowable damage limits for the Main Flap Structure of the Wing Outboard Trailing Edge shown in Wing Outboard Trailing Edge Main Flap Structure Location, Figure 101/ALLOWABLE DAMAGE 3.

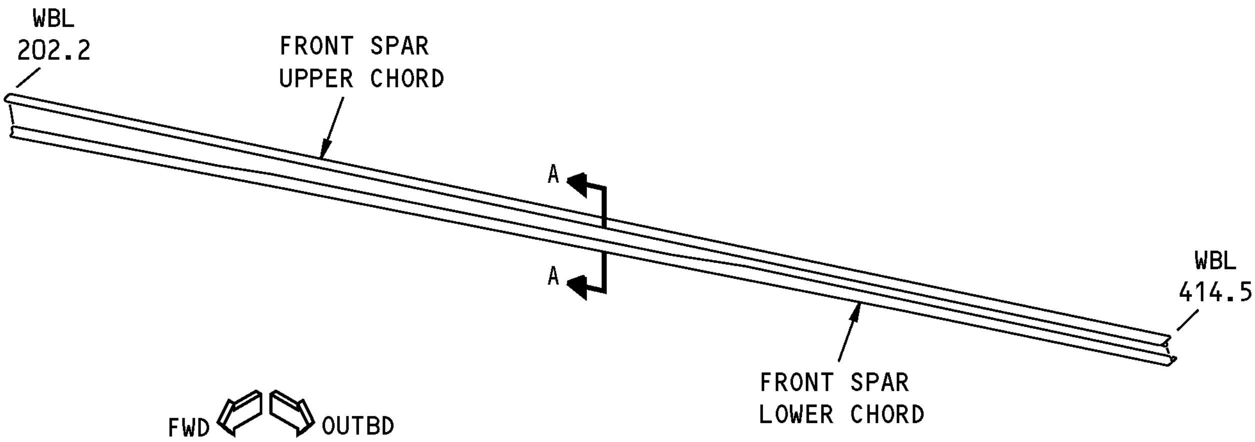



**Wing Outboard Trailing Edge Main Flap Structure Location**
**Figure 101**



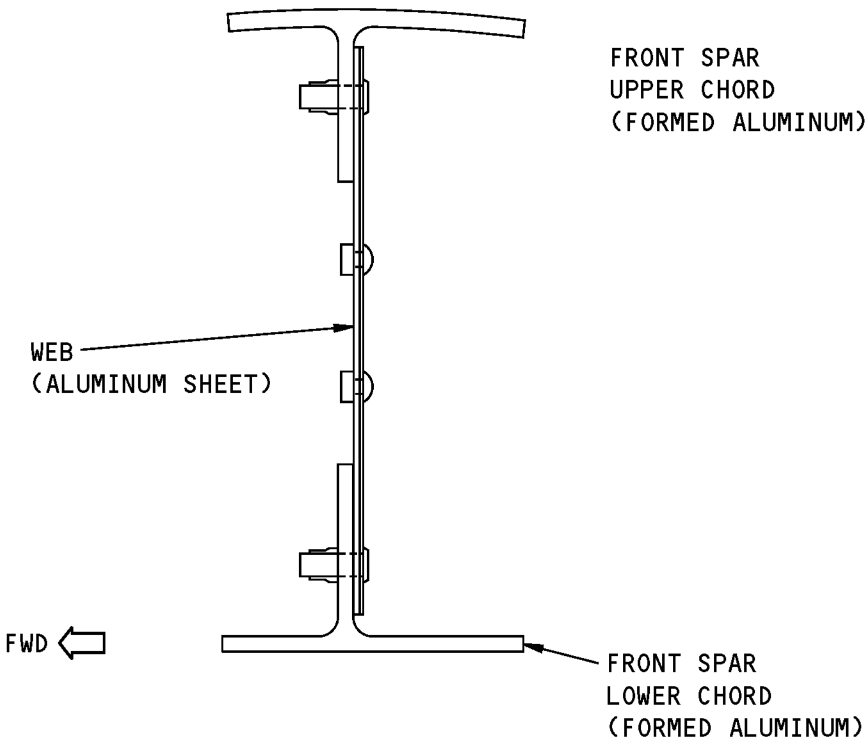

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Outboard Trailing Edge Main Flap Structure Allowable Damage**
**Figure 102 (Sheet 1 of 3)**



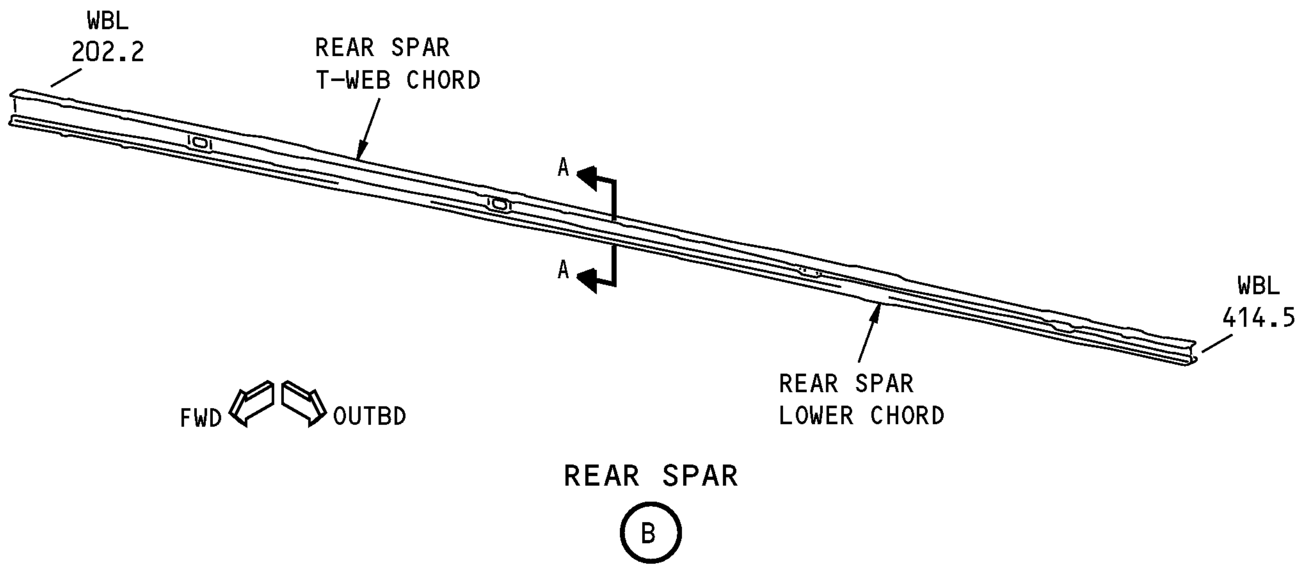

FRONT SPAR

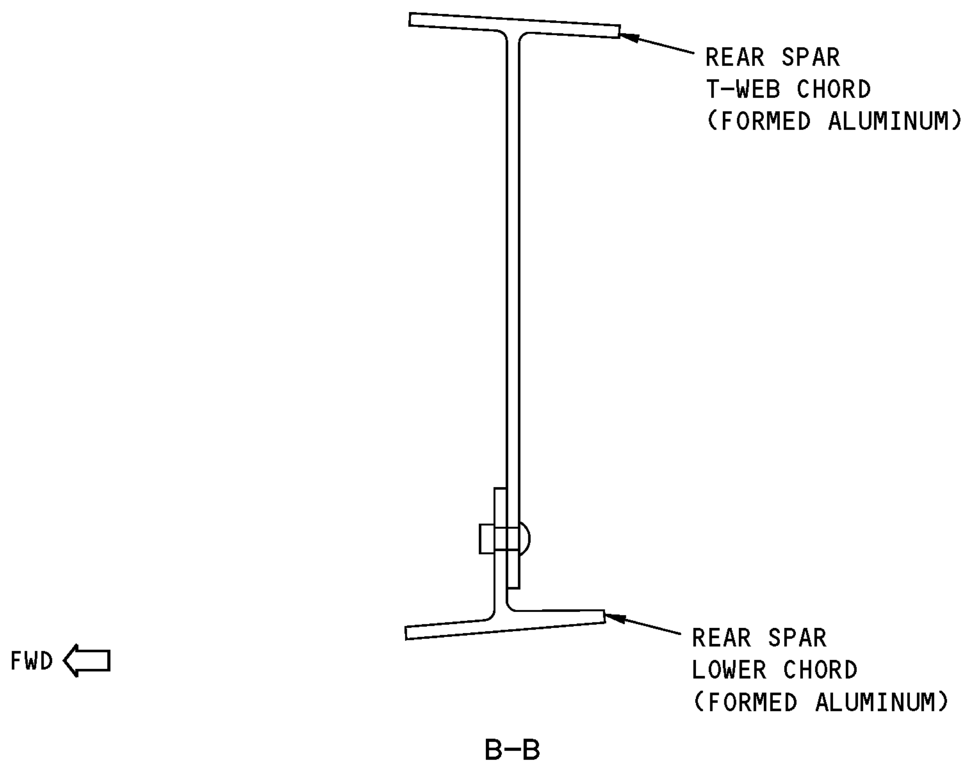
A




A-A

**Wing Outboard Trailing Edge Main Flap Structure Allowable Damage**
**Figure 102 (Sheet 2 of 3)**




**Wing Outboard Trailing Edge Main Flap Structure Allowable Damage**
**Figure 102 (Sheet 3 of 3)**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 3 for the allowable damage limits.

**WARNING:** SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STEAM EXPLOSION. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

B. Do the steps that follow if you have damage to the aluminum, titanium, or Corrosion Resistant Steel (CRES) parts:

(1) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures for the aluminum and titanium parts.

(b) Refer to SOPM 20-20-01 for the investigation and cleanup procedures for the CRES parts.

(c) Refer to 51-30-03 for possible sources of non-metallic materials you need to remove the damage.

(d) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

(2) After you remove the damage, do as follows:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(a) Flap peen or shot peen all parts but not the parts that follow:

1) Airload Rib

2) Inboard Rib

3) Cove Panel Support Rib

4) Stiffener

(b) Refer to 51-20-06 for shot peen intensity and shot number.

(c) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

(d) Apply the solvent resistant chemicals.

1) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

2) Apply cadmium plating to the bare surfaces of the CRES parts. Refer to 51-20-01.

(e) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked CRES parts. Refer to SOPM 20-41-02.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| **TYPE OF STRUCTURE** | **PARAGRAPH** |
| STIFFENER AND SPAR CHORDS | 4.A |

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | |
|---|---|
| TYPE OF STRUCTURE | PARAGRAPH |
| FRONT SPAR WEB | 4.B |
| RIB SUPPORTS | 4.C |

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-01 | Magnetic Particle Inspection |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Front and Rear Spar Chords, and Stiffener

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , and F.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , F , G , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Front Spar Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 3, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 3, Details A , B , E , G , and H.

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 3, Detail I.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 2.0 inches away from a fastener hole, the edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet installed without sealant.

C. Ribs, Rib Supports, and Fittings

(1) Cracks:

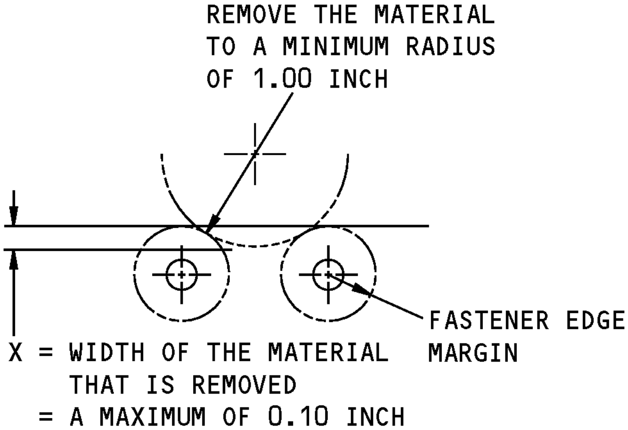
(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , and F.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , F , G , and H.

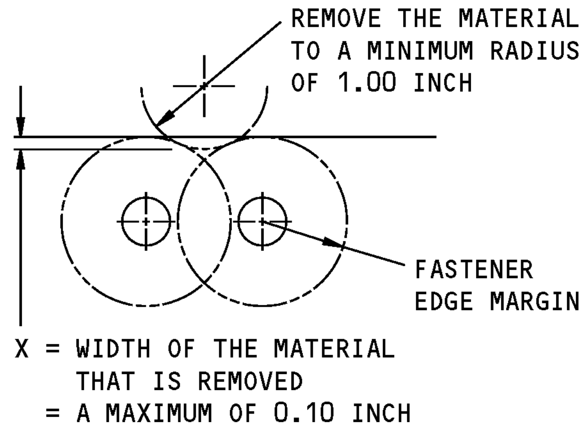
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



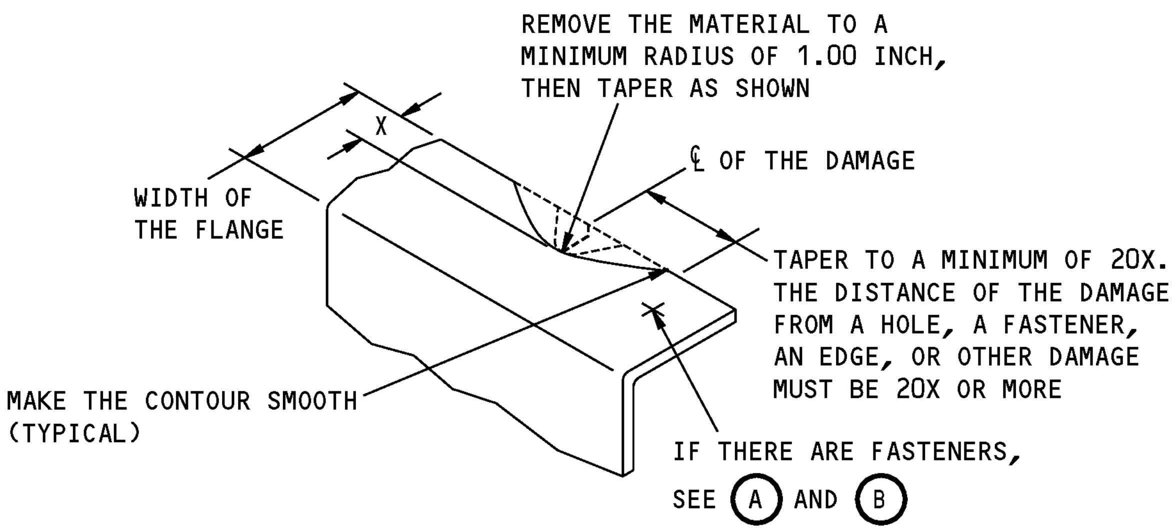

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

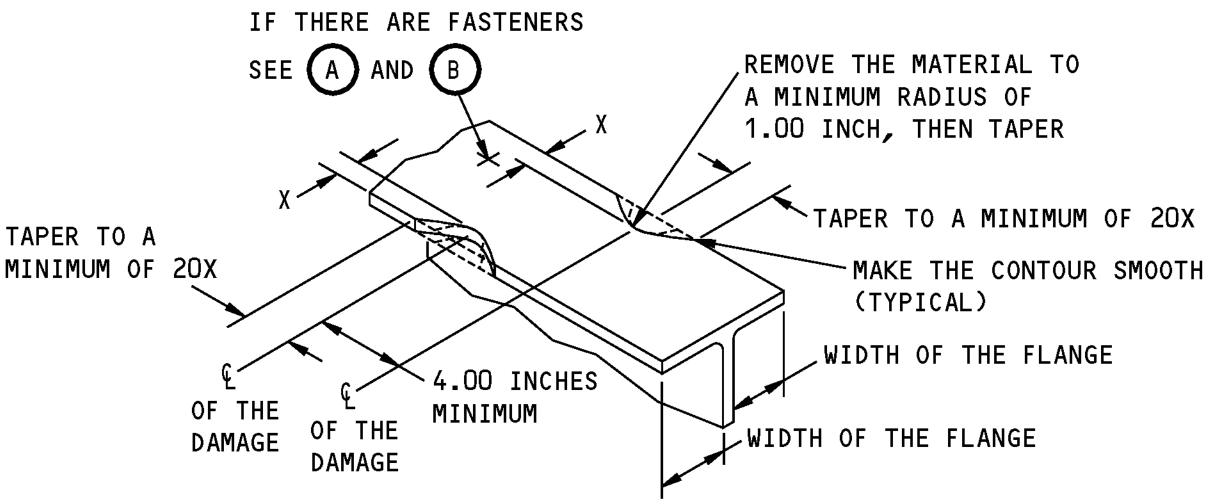



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

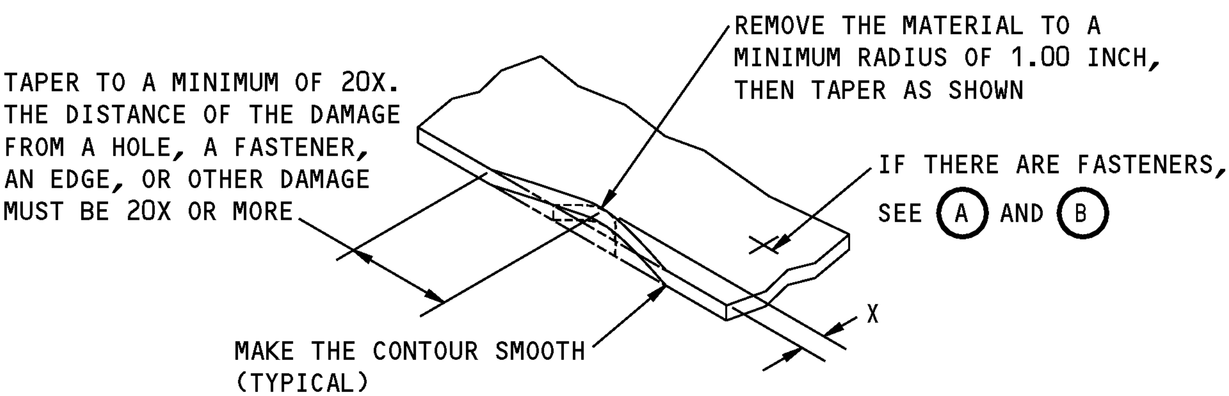
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 4)**




X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

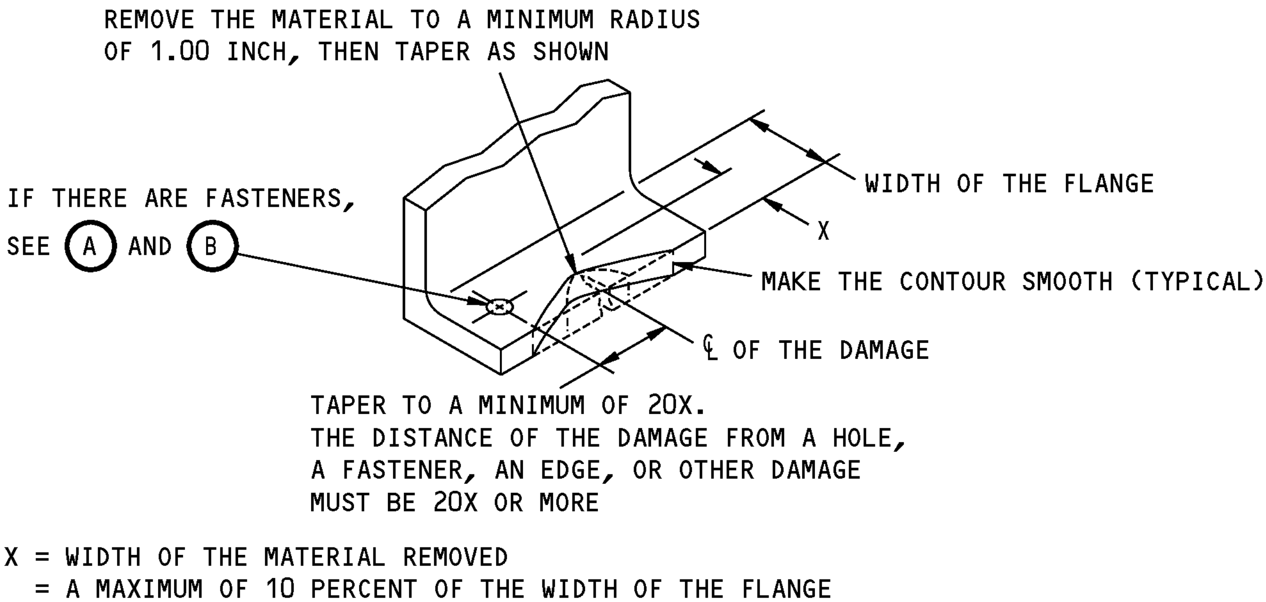



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

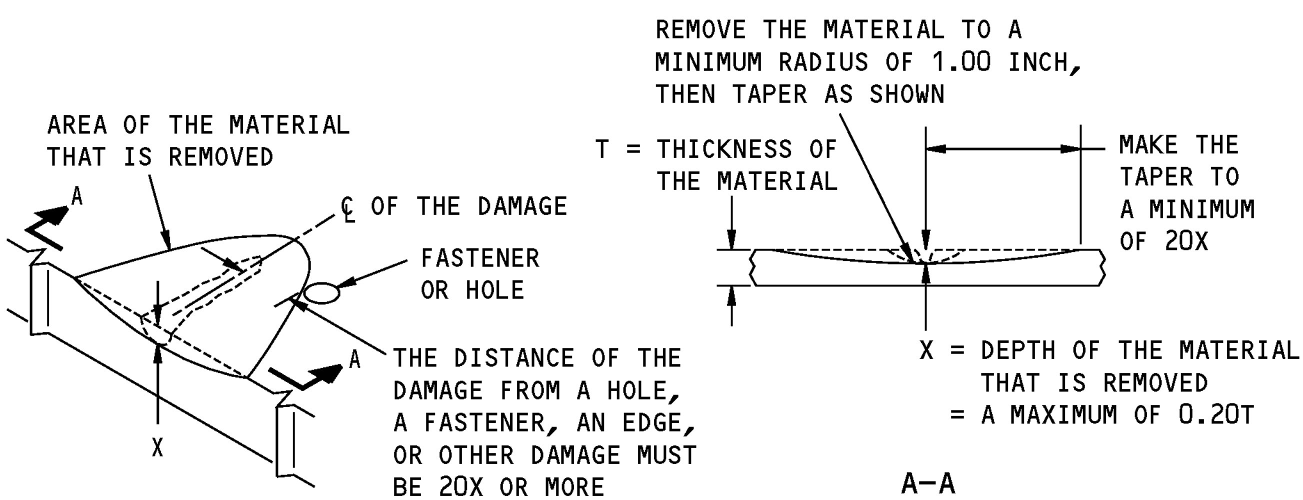
E

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 4)**




REMOVAL OF DAMAGED MATERIAL AT AN EDGE

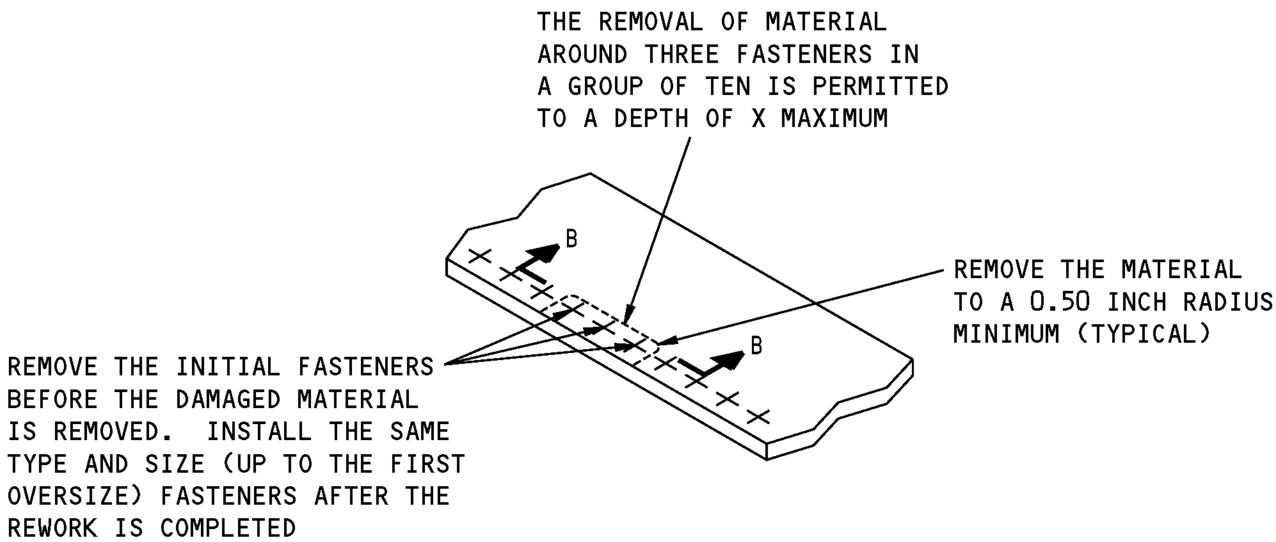
F




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

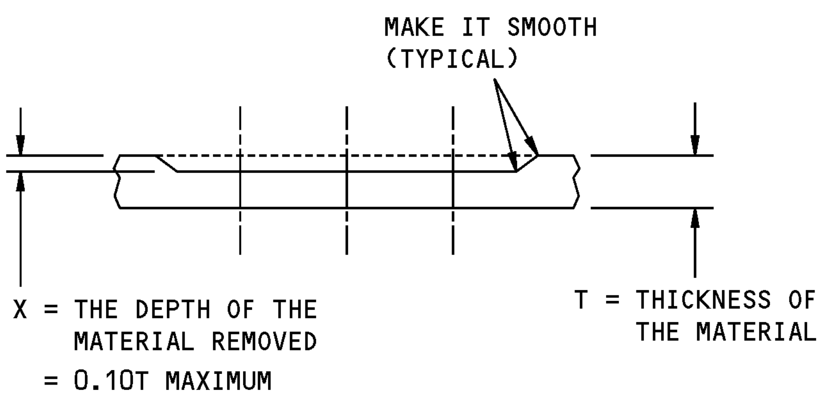
G

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 4)**




REMOVAL OF CORROSION AROUND THE FASTENERS

H



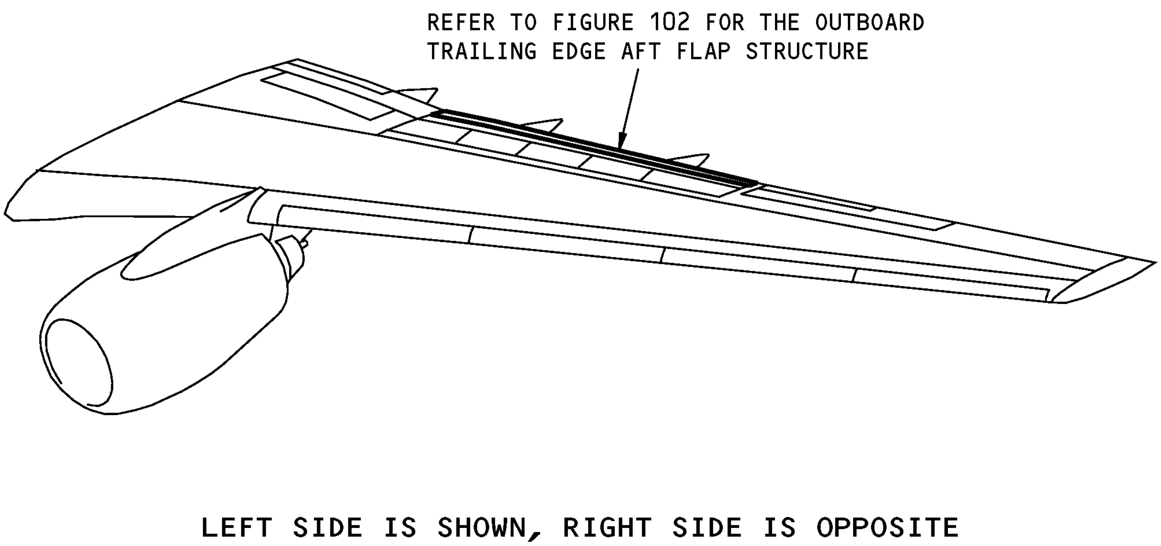

B-B

**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 4)**

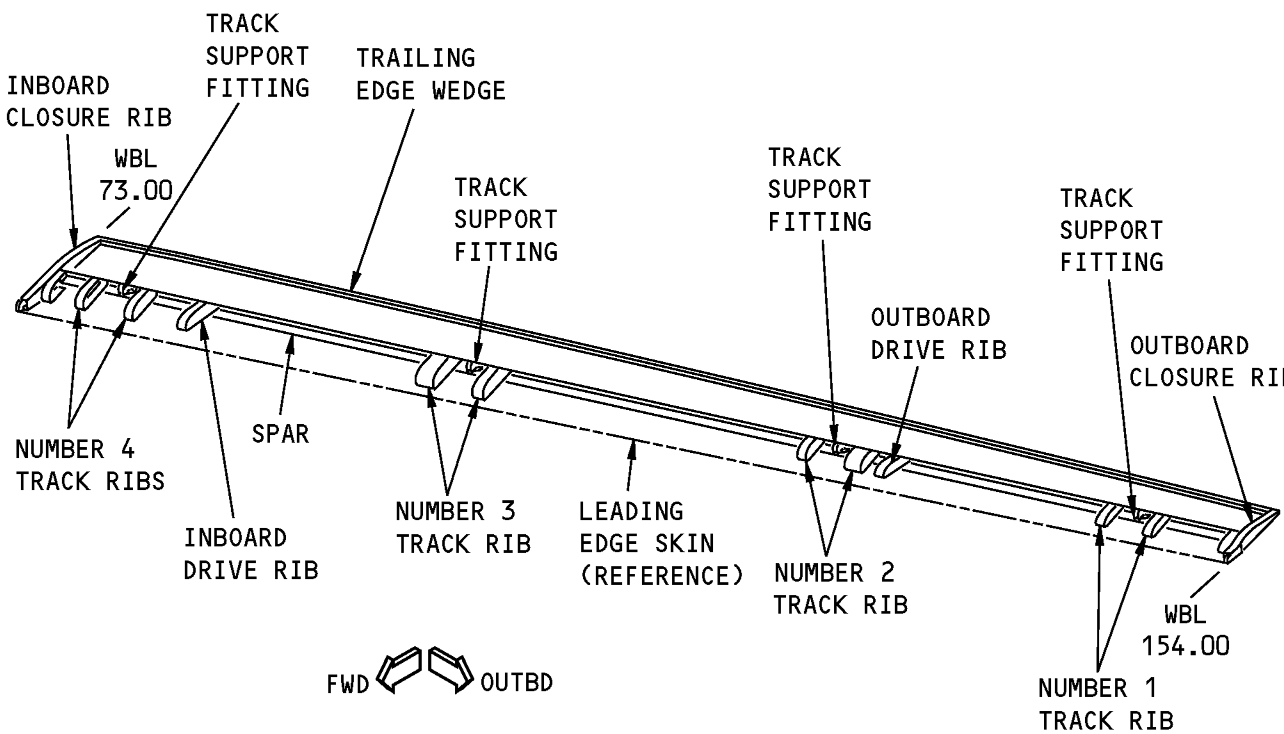
## ALLOWABLE DAMAGE 4 - WING OUTBOARD TRAILING EDGE AFT FLAP STRUCTURE

### 1. Applicability

A. This subject gives the allowable damage limits for the Aft Flap Structure of the Wing Outboard Trailing Edge shown in Wing Outboard Trailing Edge Aft Flap Location, Figure 101/ALLOWABLE DAMAGE 4.




**Wing Outboard Trailing Edge Aft Flap Location**
**Figure 101**




NOTE: ALL PARTS IDENTIFIED ARE MADE OF ALUMINUM.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD TRAILING EDGE AFT FLAP STRUCTURE

**Wing Outboard Trailing Edge - Aft Flap Structure Location**
**Figure 102**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 4 for the allowable damage limits.

B. Do the steps that follow if you have damage.

(1) Remove the damage. Refer to 51-10-02 for the investigation and cleanup procedures.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(4) Make the edges of the blendouts and holes smooth to a finish of 125 microinches Ra or better.

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(5) Flap peen or shot peen the reworked areas of the track support fittings, but not the inner surfaces of the lug bores.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(6) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(7) Apply a layer of BMS 10-11, Type I primer to the reworked surfaces. Refer to SOPM 20-41-02.

3. **References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. **Allowable Damage Limits**

A. Track Ribs, Drive Ribs, and Spar

(1) Rib and Spar Webs

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail E .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details E , G , and H .

(c) Dents:

1) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail I .

**NOTE**: Fill the space between the flange and skin with BMS 5-95 sealant if the space is more than 0.005 inch.

(d) Holes and Punctures:

1) The damage is permitted if it is:

a) A maximum of 0.25 inch in diameter

b) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage

c) Filled with a 2117-T3 or 2117-T4 protruding head rivet installed without sealant.

(2) Rib and Spar Chords

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , C , D , and F.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , C , D , F , G , and H .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

B. Inboard Closure Rib, Outboard Closure Rib, and Fittings

(1) Cracks:

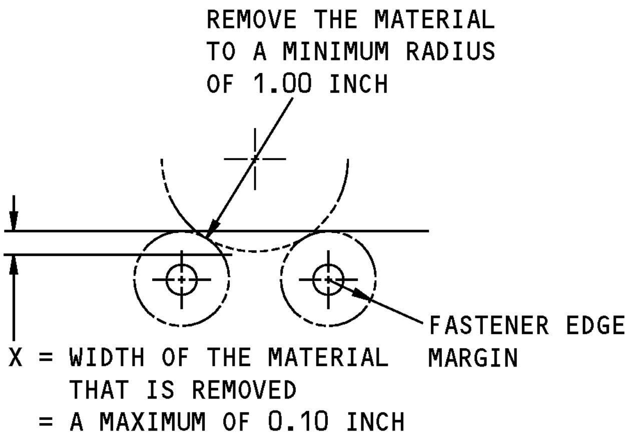
(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 4, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 4, Details A , B , C , D , and E .

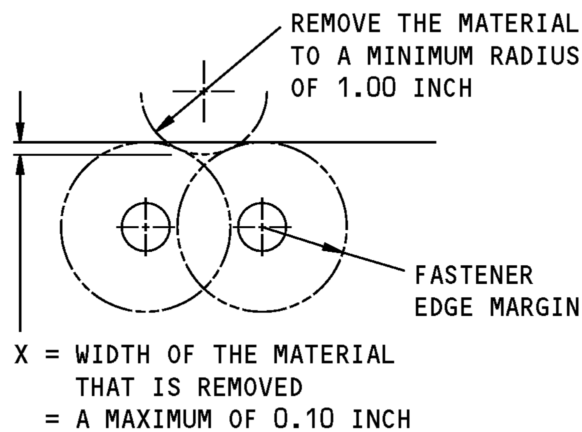
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



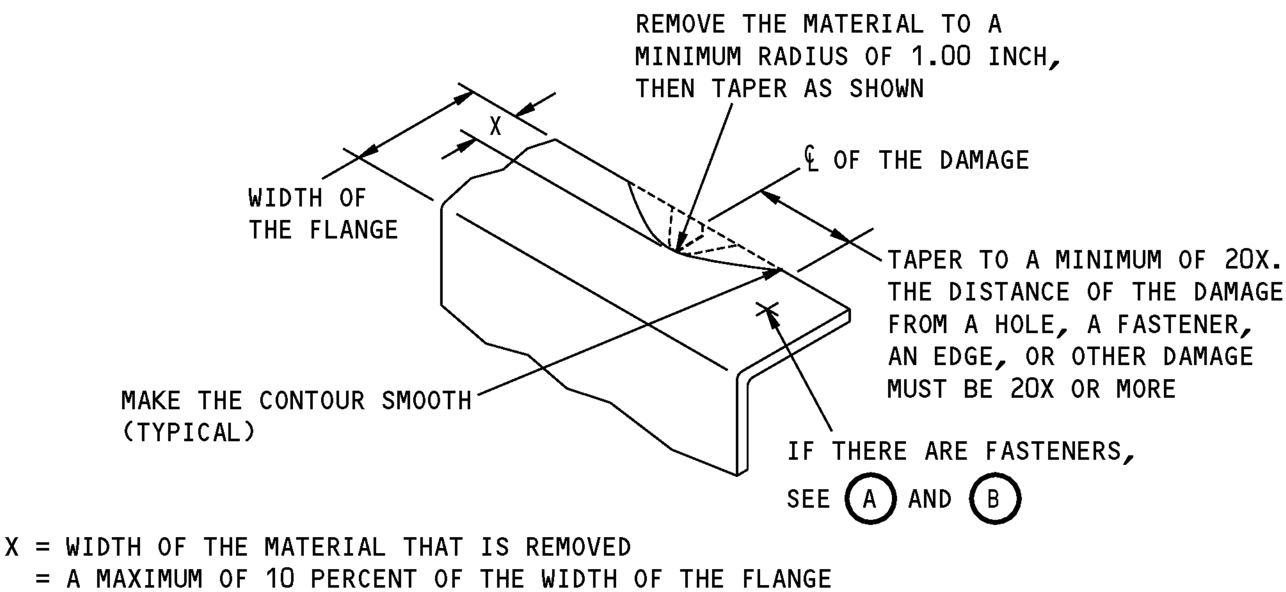

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

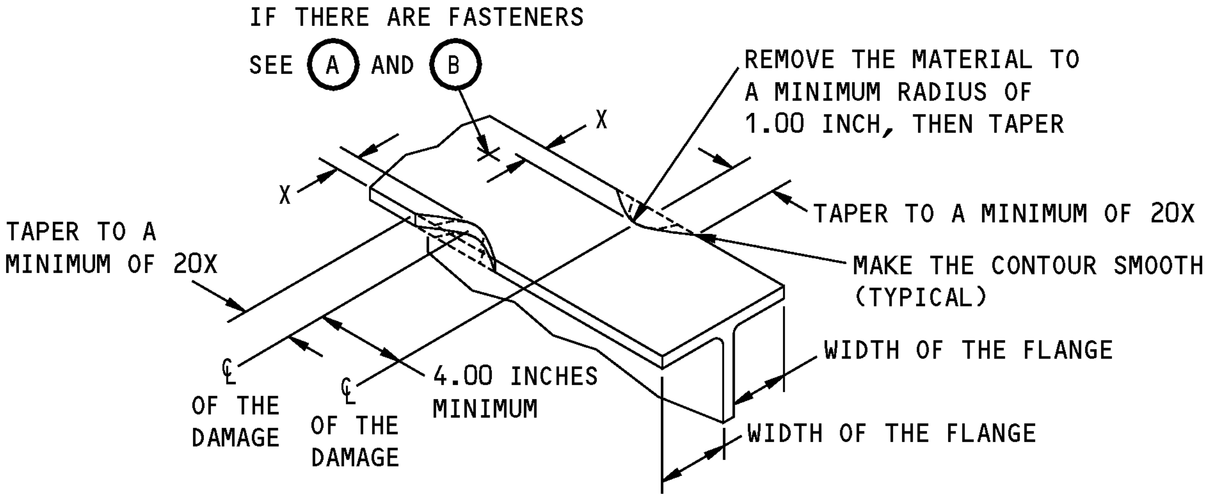
B




REMOVAL OF DAMAGED MATERIAL ON AN EDGE OF A FORMED PART

C

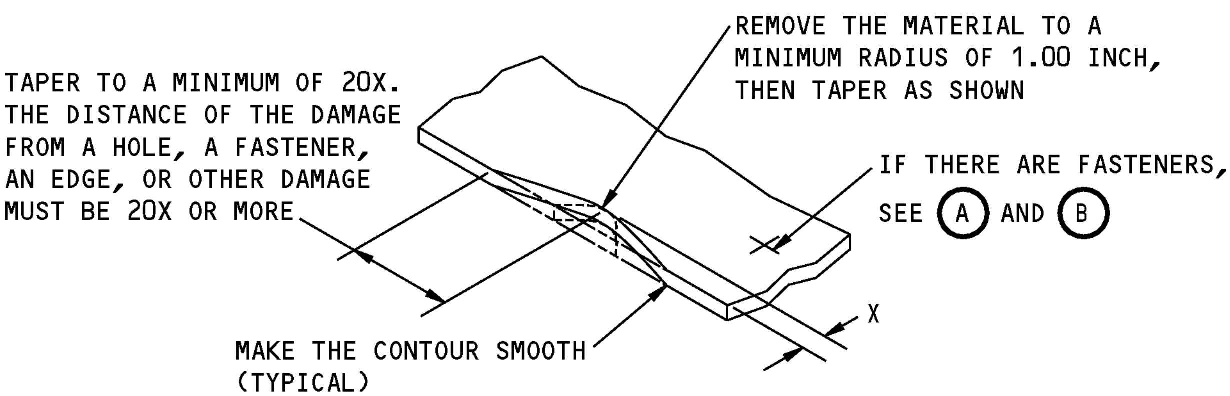
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 4)**




X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

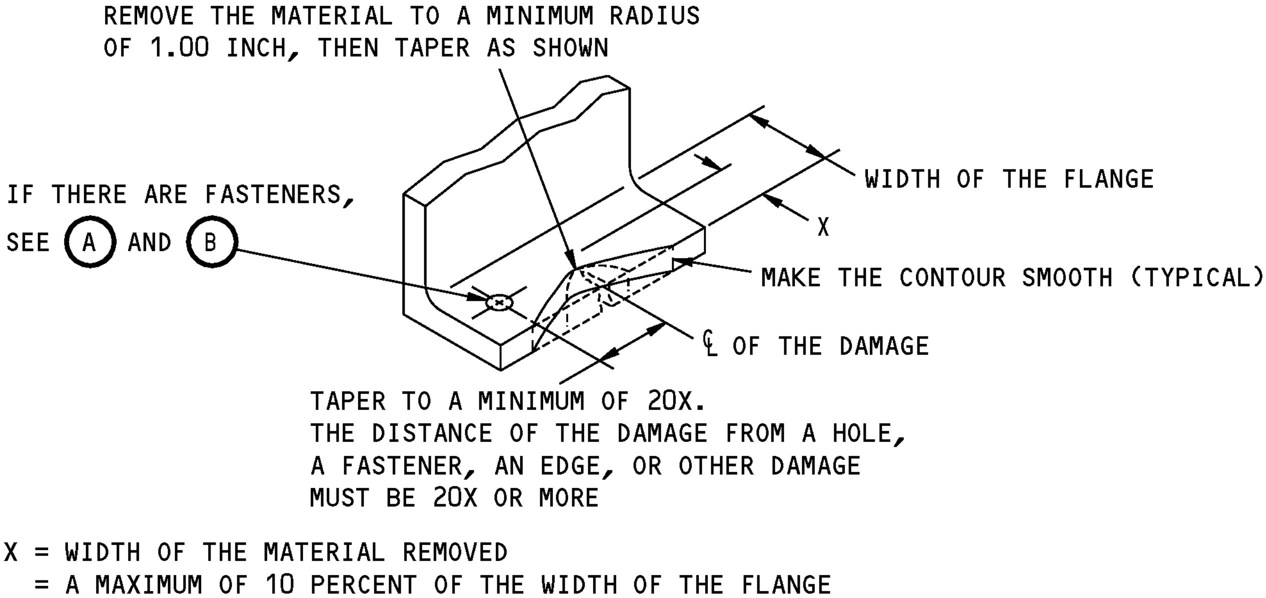



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

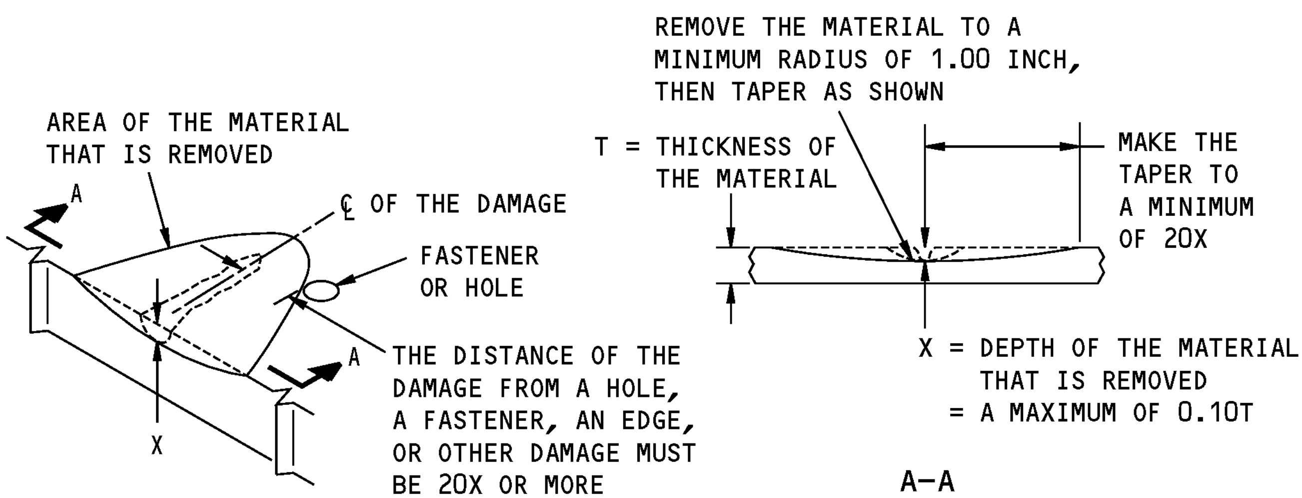
E

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 4)**




REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF AN EXTRUDED PART

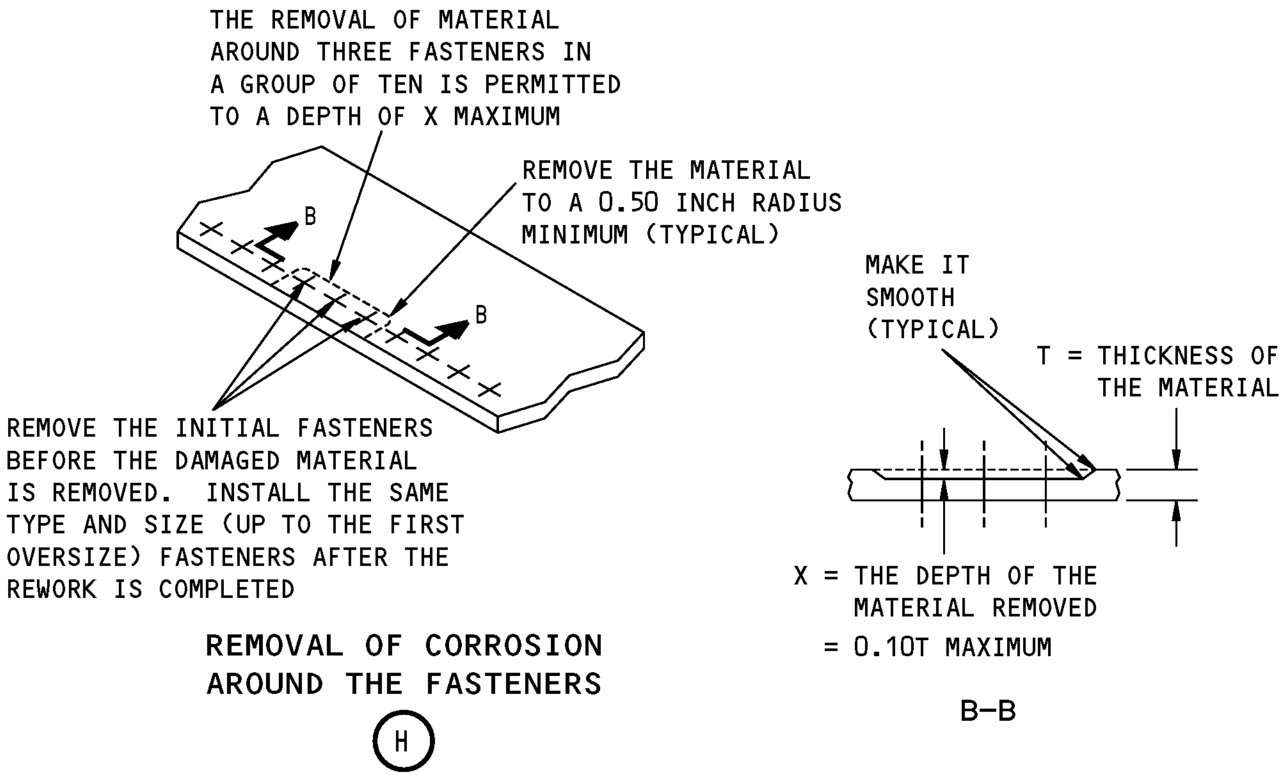
F

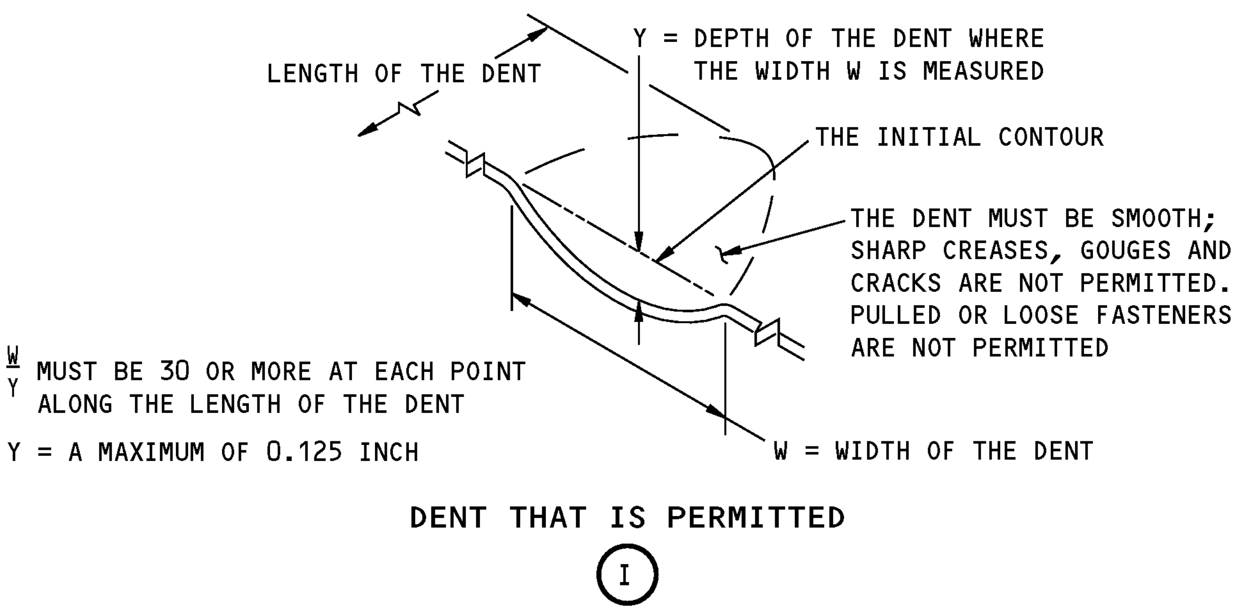



REMOVAL OF DAMAGED MATERIAL ON A SURFACE

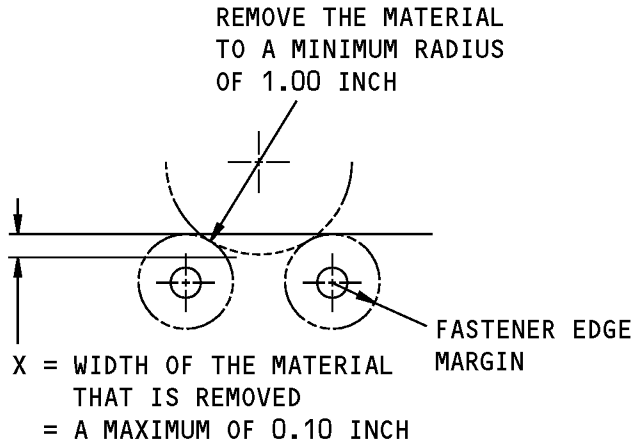
G

**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 4)**



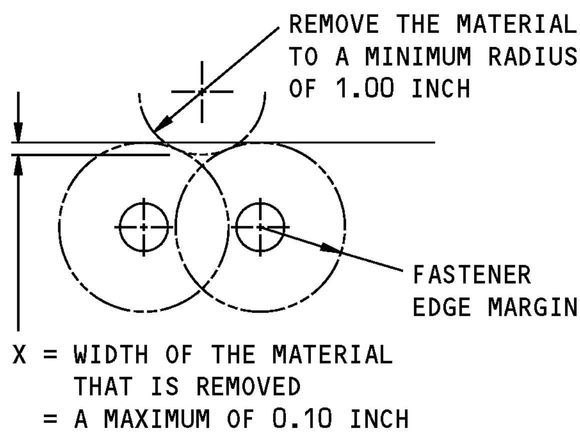




**Allowable Damage Limits**
**Figure 103 (Sheet 4 of 4)**




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

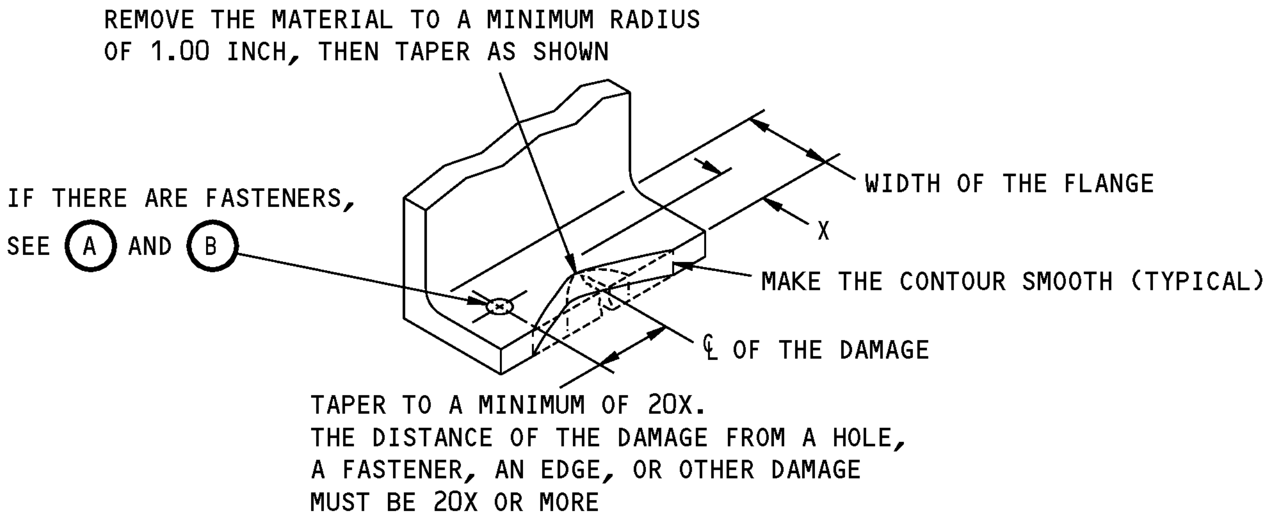
A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

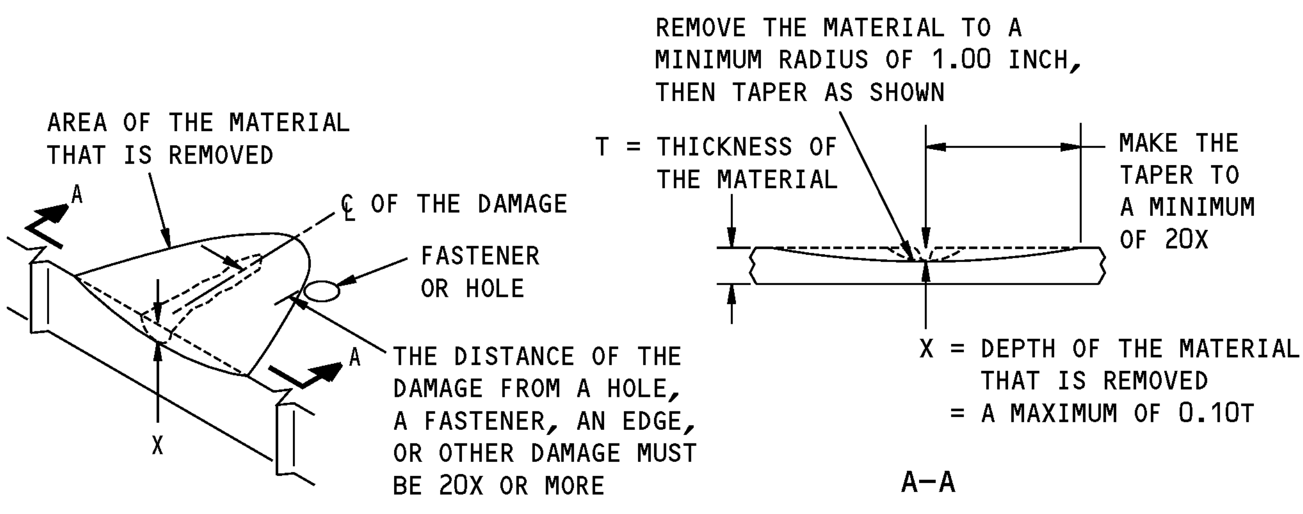
**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**




X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

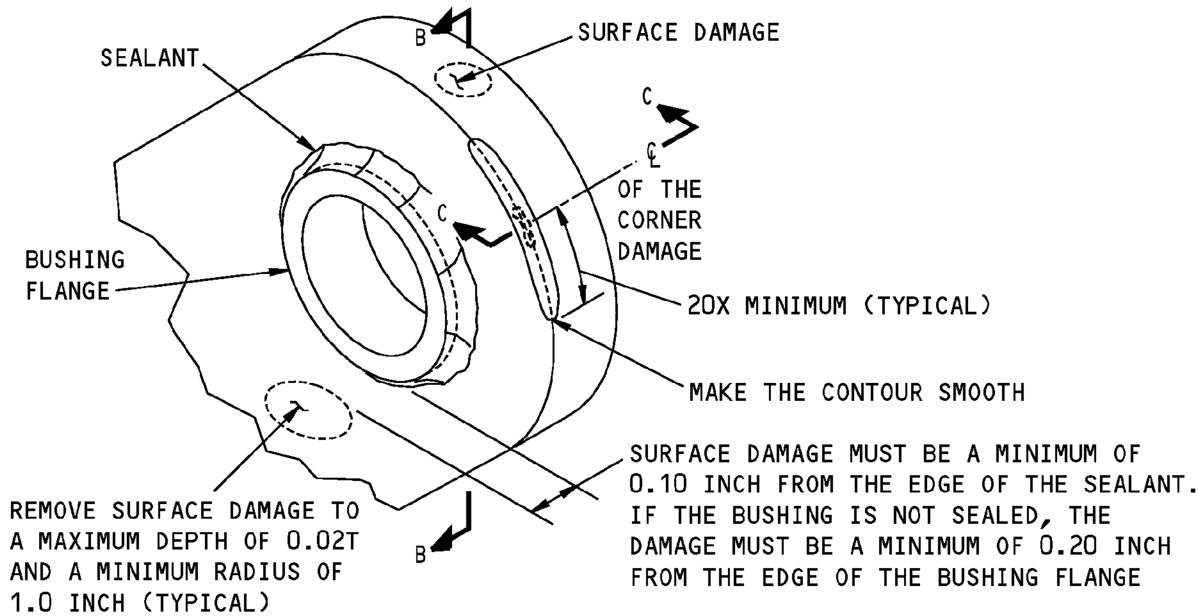
C




REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

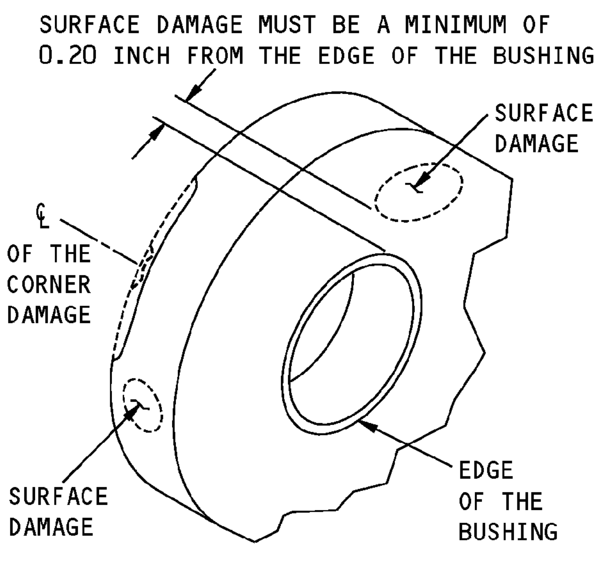
**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**



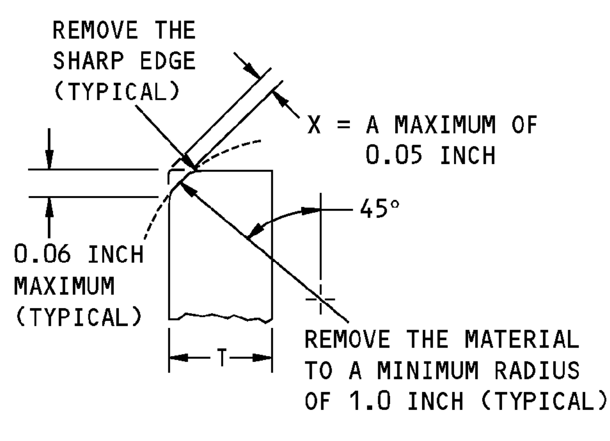

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

E



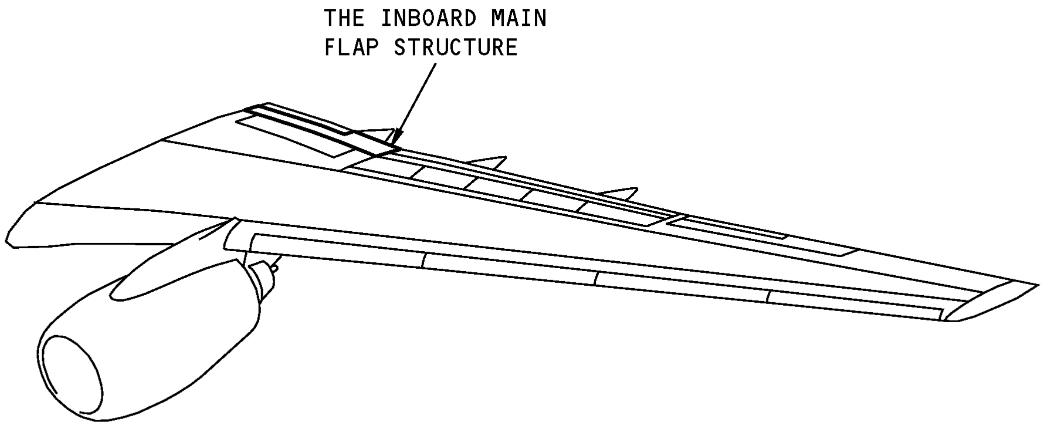

B-B




C-C

**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**

## REPAIR 1 - WING INBOARD TRAILING EDGE MAIN FLAP STRUCTURE

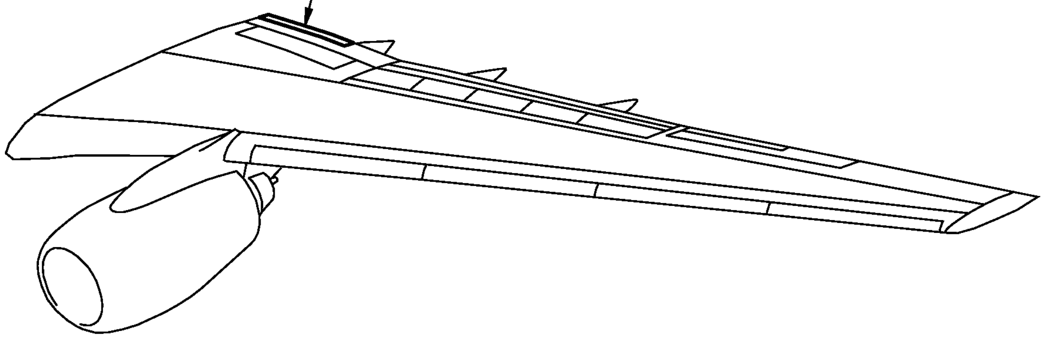



NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Inboard Trailing Edge Main Flap Location**
**Figure 201**

## REPAIR 2 - WING INBOARD TRAILING EDGE AFT FLAP STRUCTURE

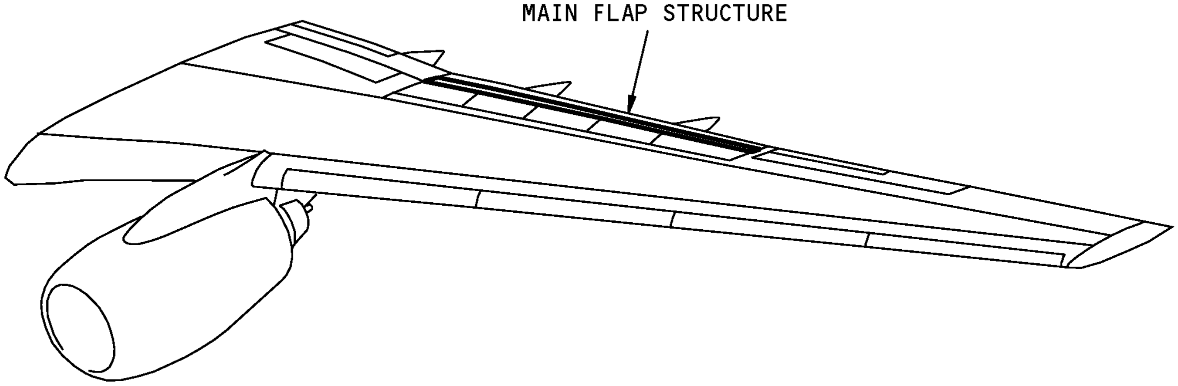



NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Inboard Trailing Edge Aft Flap Location**
**Figure 201**

## REPAIR 3 - WING OUTBOARD TRAILING EDGE MAIN FLAP STRUCTURE

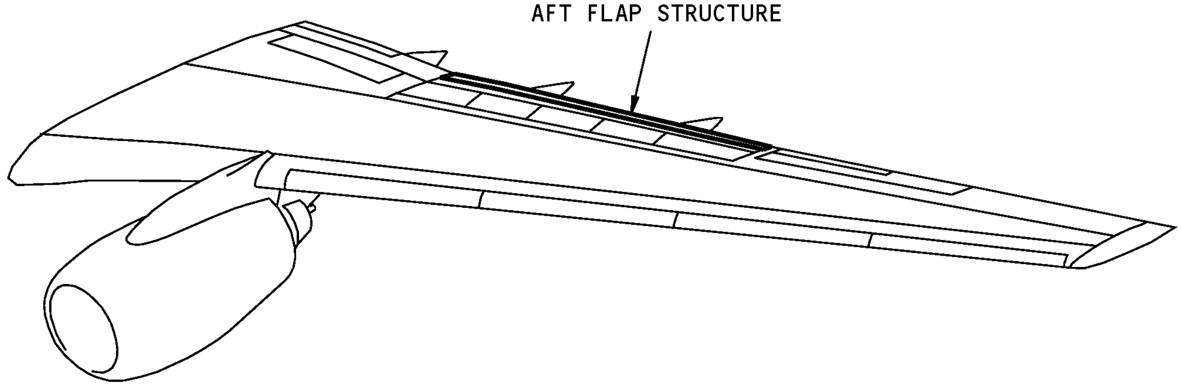



NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Outboard Trailing Edge Main Flap Location**
**Figure 201**

## REPAIR 4 - WING OUTBOARD TRAILING EDGE AFT FLAP STRUCTURE




NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Outboard Trailing Edge Aft Flap Location**
**Figure 201**

## REPAIR 5 - INBOARD FLAP AFT FLAP TRACK OVERSIZED BEARING BORE REPAIR

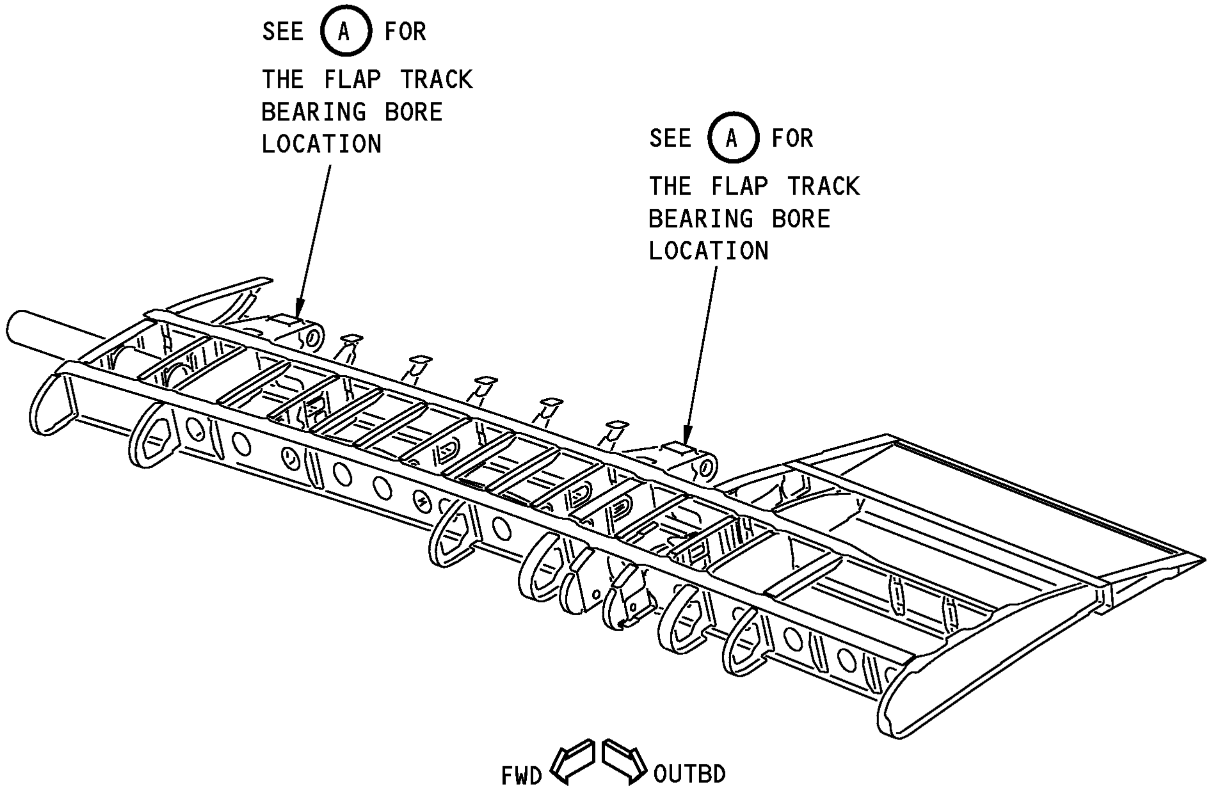
1. **Applicability**

   A. This repair is applicable for oversized bearing bores in the titanium inboard flap aft flap track with part numbers 113A2672–3 and 113A2672–7.

   B. Do not use this repair if the wear in the lug bore is more than 1.000 in. (25.40 mm) in diameter.

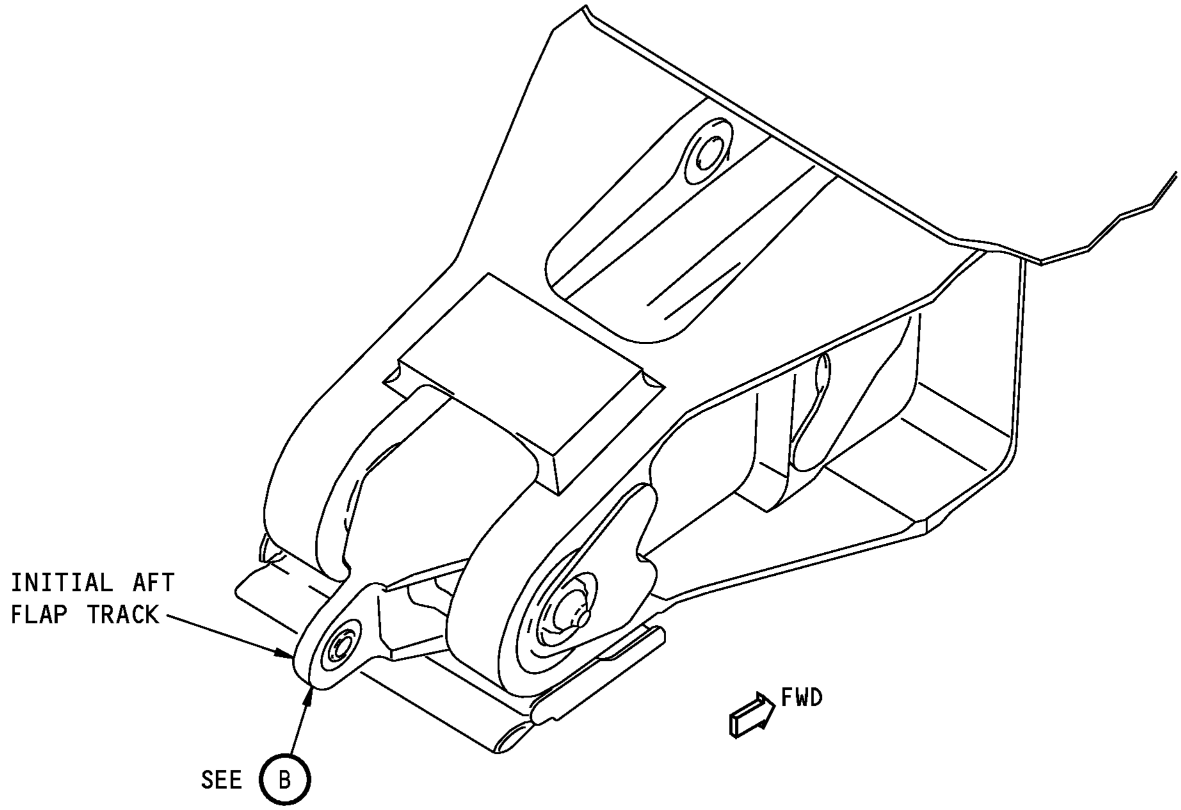
2. **General**

   A. For the location of the repair area, see Figure 201/REPAIR 5.



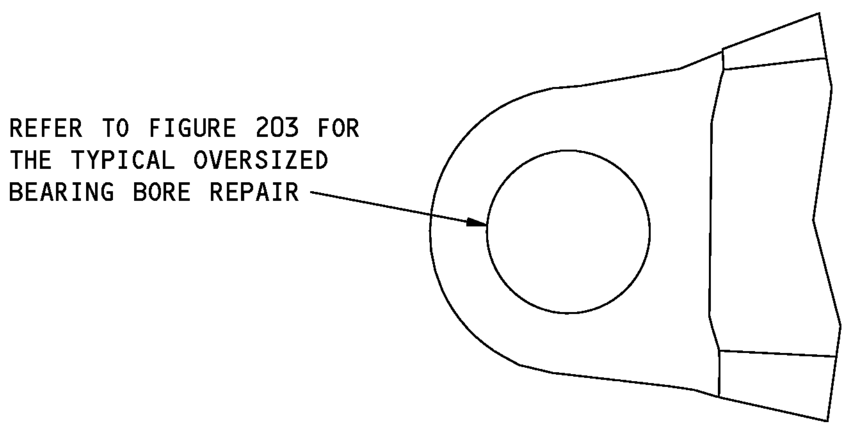

1856210 S0000331818_V1

**Location of the Inboard Flap Aft Flap Track Bushing Bore**
**Figure 201 (Sheet 1 of 2)**




AFT FLAP TRACK BEARING BORE LOCATION

A




VIEW IN THE OUTBOARD DIRECTION

B

1857827 S0000331819_V1

**Location of the Inboard Flap Aft Flap Track Bushing Bore**
**Figure 201 (Sheet 2 of 2)**

3. **References**

| Reference | Title |
|---|---|
| AMM 51-21 | INTERIOR AND EXTERIOR FINISHES |
| BAC 5619 | Heat Treat of Corrosion Resistant Steel |
| BAC 5625 | Boeing Aircraft Company Process Specification |
| CMM 57-53-05 | Inboard Flap - Main Flap Final Assembly |
| SOPM 20-50-03 | Bearing and Bushing Replacement |

4. **Repair Instructions**

A. Get access to the damaged area.

B. Remove the initial bearing. Refer to SOPM 20-50-03.

C. Ream the lug bore to a maximum of 1.000 in. (25.40 mm) in diameter.

D. Chamfer the inboard and outboard edges of the lug bore. Refer to Figure 203/REPAIR 5

E. Make the Part [1] Repair Sleeve. See Table 201/REPAIR 5

(1) The outer diameter of the Part [1] Repair Sleeve must be between 0.0007 in. (0.018 mm) and 0.0020 in. (0.051 mm) larger than the reamed bore diameter.

(2) The wall thickness of the Part [1] Repair Sleeve must be a minimum of 0.040 in. (1.02 mm) thick after you do the inner diameter ream.

(3) Heat treat the Part [1] Repair Sleeve. Refer to BAC 5619.

(a) If you make the Part [1] Repair Sleeve out of 15–5PH, heat treat the sleeve to between 140 KSI and 160 KSI.

(b) If you make the Part [1] Repair Sleeve out of 17–4PH, heat treat the sleeve to between 150 KSI and 170 KSI.

(4) If you install this repair on a 113A2672–3 flap track, you must do the steps below. If you install this repair on a 113A2672–7 flap track, do not do these steps.

(a) Make a 0.090 in. (2.29 mm) wide and 0.015 in. (0.38 mm) deep grease groove in the outer diameter of the Part [1] Repair Sleeve. See Figure 203/REPAIR 5.

**NOTE**: The grease groove in the Part [1] Repair Sleeve must align with the grease groove in the Part [2] Bearing.

(b) Make four 0.090 in. (2.29 mm) diameter holes through the grease groove to make a grease path to the bearing. Make sure that there is an equal distance between all the holes. See Figure 203/REPAIR 5.

**Table 201**: Repair Materials

| ITEM | PART | QUANTITY | MATERIAL |
|---|---|---|---|
| [1] | Repair Sleeve | 1 | Use 15–5PH steel per AMS 5659, or use 17–4PH steel per AMS 5643 |
| [2] | Bearing | 1 | Same size and type as the initial bearing. |

F. Remove all nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Repair Sleeve and the lug bore repair area.

G. Make the surfaces of the Part [1] Repair Sleeve and the lug bore repair area smooth to a finish of 63 microinches RA or smoother.

H. Passivate the Part [1] Repair Sleeve. Refer to Method II of BAC 5625.

**WARNING:** WHEN YOU DO WORK WITH DRY ICE, MAKE SURE YOU USE SAFETY GOGGLES AND INSULATED GLOVES. IF DRY ICE TOUCHES YOUR EYES OR SKIN, FROSTBITE CAN OCCUR.

**WARNING:** LIQUID NITROGEN IS AN AGENT THAT IS AN ASPHYXIANT. MAKE SURE ALL PERSONS OBEY ALL OF THE PRECAUTIONS WHEN LIQUID NITROGEN IS USED.

- USE IN AN AREA OPEN TO THE AIR.
- CLOSE THE CONTAINER WHEN NOT USED.
- DO NOT GET LIQUID NITROGEN IN THE EYES, ON THE SKIN, OR ON YOUR CLOTHES.
- DO NOT BREATHE THE GAS.

I. Install the Part [1] Repair Sleeve. Use a shrink fit method. Install the Part [1] Repair Sleeve with BMS 3-33. Refer to SOPM 20-50-03, Section 7, paragraph B.

J. Swage the Part [1] Repair Sleeve on both the inboard and outboard ends. See Figure 203/REPAIR 5.

K. Ream the inner diameter of the Part [1] Repair Sleeve to the necessary diameter for the Part [2] Bearing.

L. Chamfer the inboard and outboard edges of the Part [1] Repair Sleeve to the necessary dimensions for the Part [2] Bearing. See Figure 203/REPAIR 5.

M. Install the Part [2] Bearing. Refer to CMM 57-53-05.

N. Do a push-out load test on the bearing. Refer to SOPM 20-50-03.

O. Refinish the repair area around the lug bore if necessary. Refer to AMM SECTION 51-21.

P. Replace the items you removed for access.

NOTES

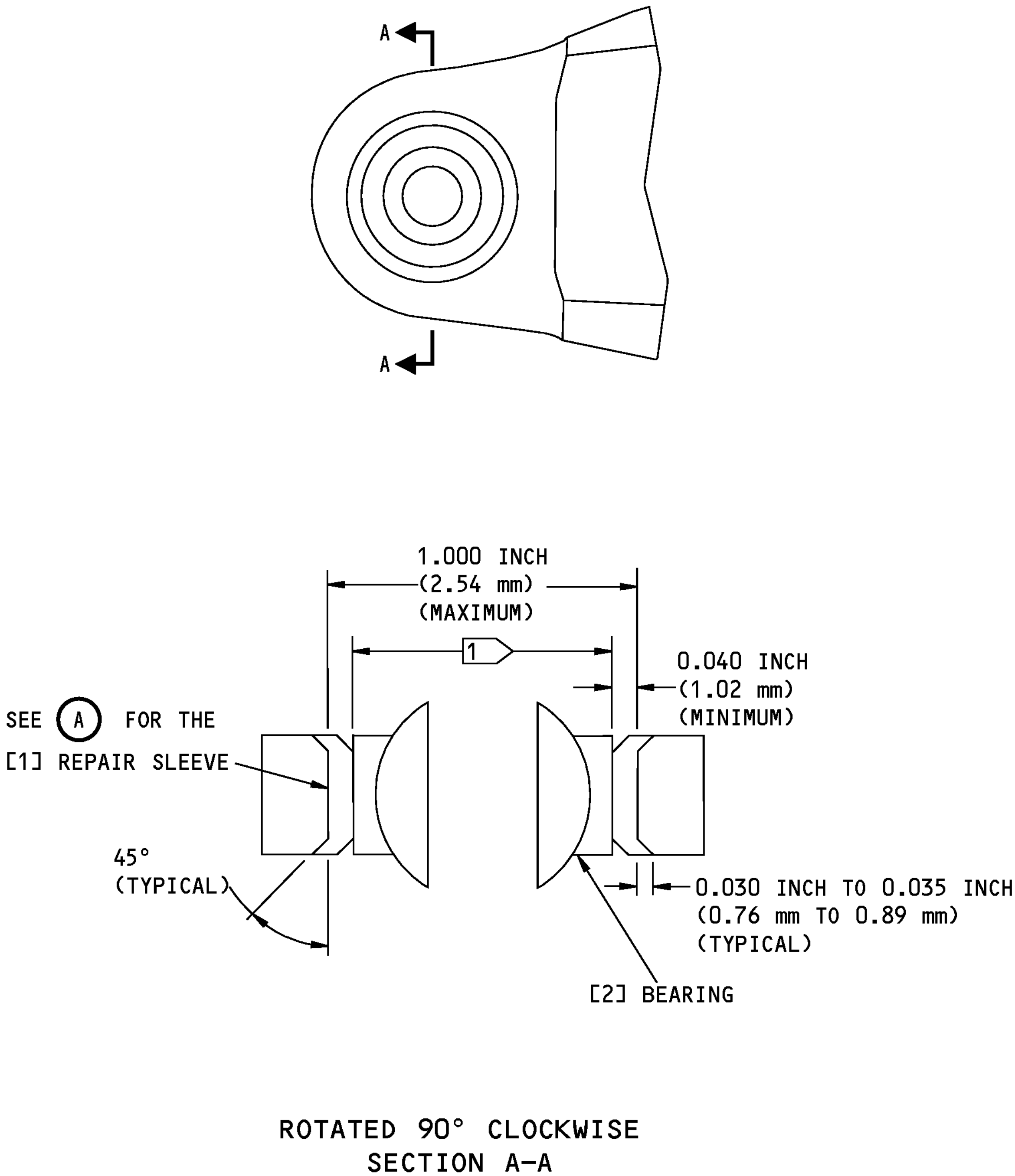
- ALL DIMENSIONS ARE INCHES (mm)

1 THE NOMINAL DIAMETER OF THE BEARING RING IS 0.7818 INCH (19.858 mm).

2 ALIGN THE GREASE GROOVE OF THE [1] REPAIR SLEEVE WITH THE GREASE GROOVE OF THE [2] BEARING.

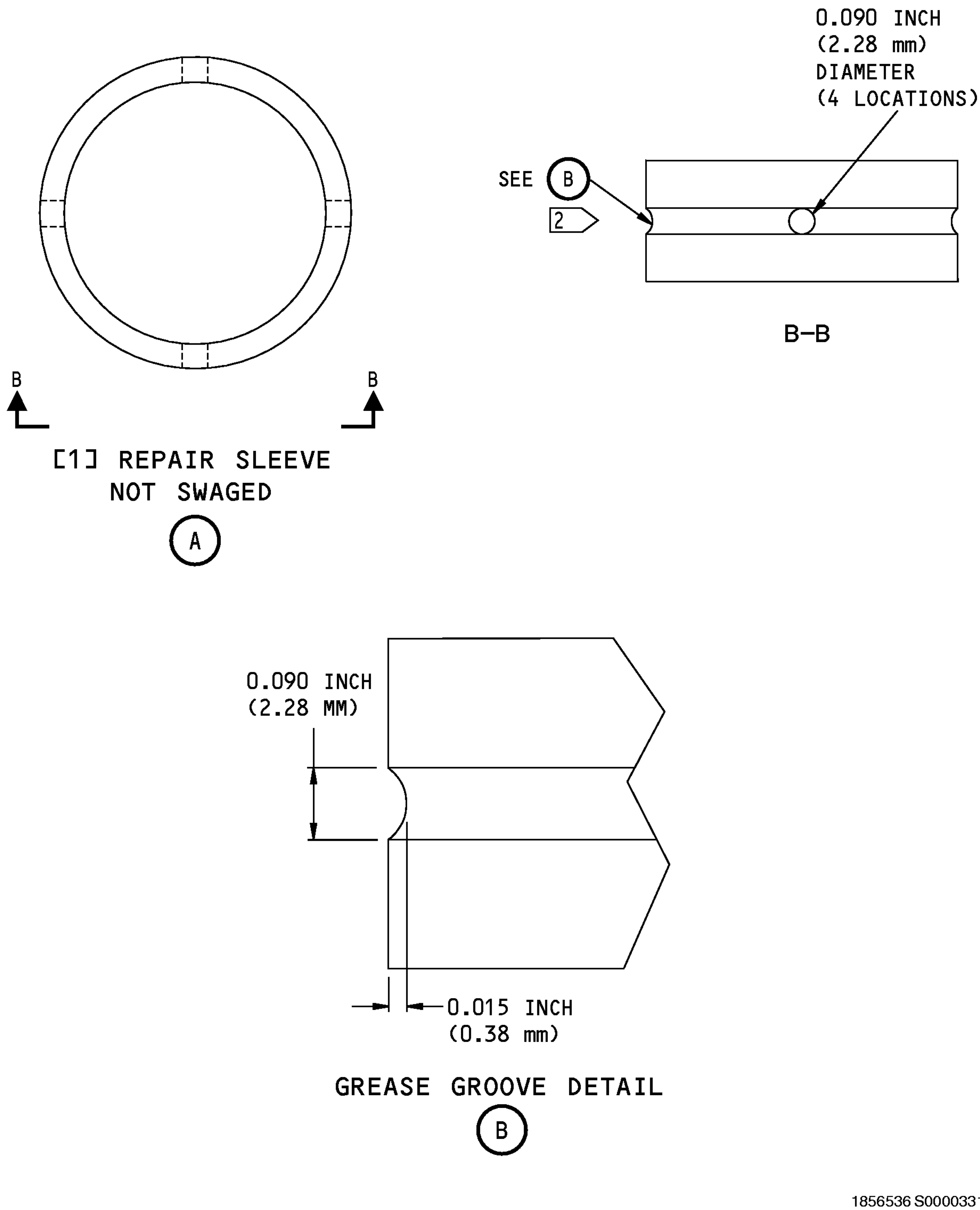
1856688 S0000331820_V1

**Notes and Fastener Symbols**
**Figure 202**



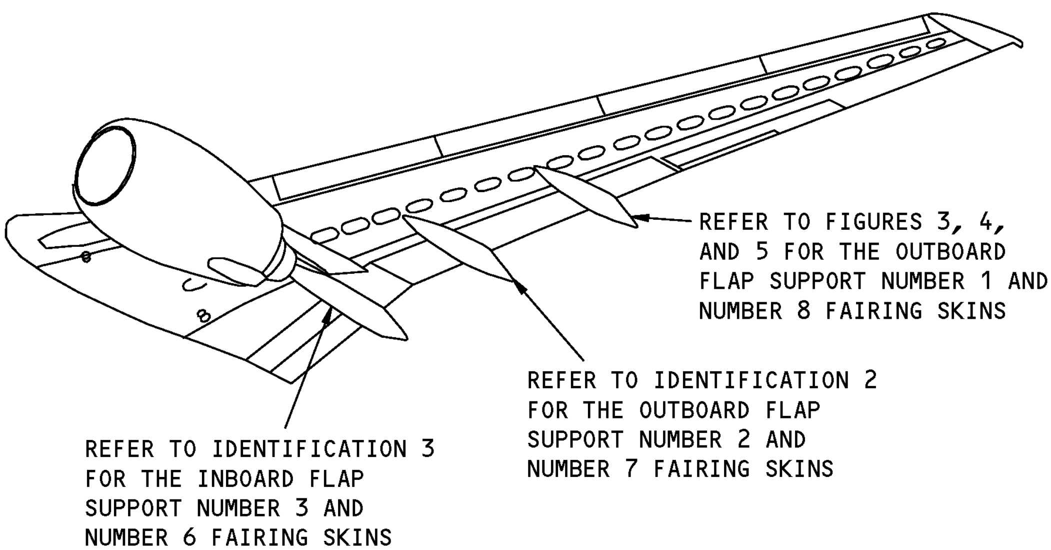

1856623 S0000331823_V1

**Repair of the Inboard Flap Aft Flap Track Oversized Bearing Bore**
**Figure 203 (Sheet 1 of 2)**




**Repair of the Inboard Flap Aft Flap Track Oversized Bearing Bore**
**Figure 203 (Sheet 2 of 2)**

## IDENTIFICATION 1 - FAIRING SKINS - WING OUTBOARD TRAILING EDGE FLAP SUPPORTS - NUMBERS 1 AND 8




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

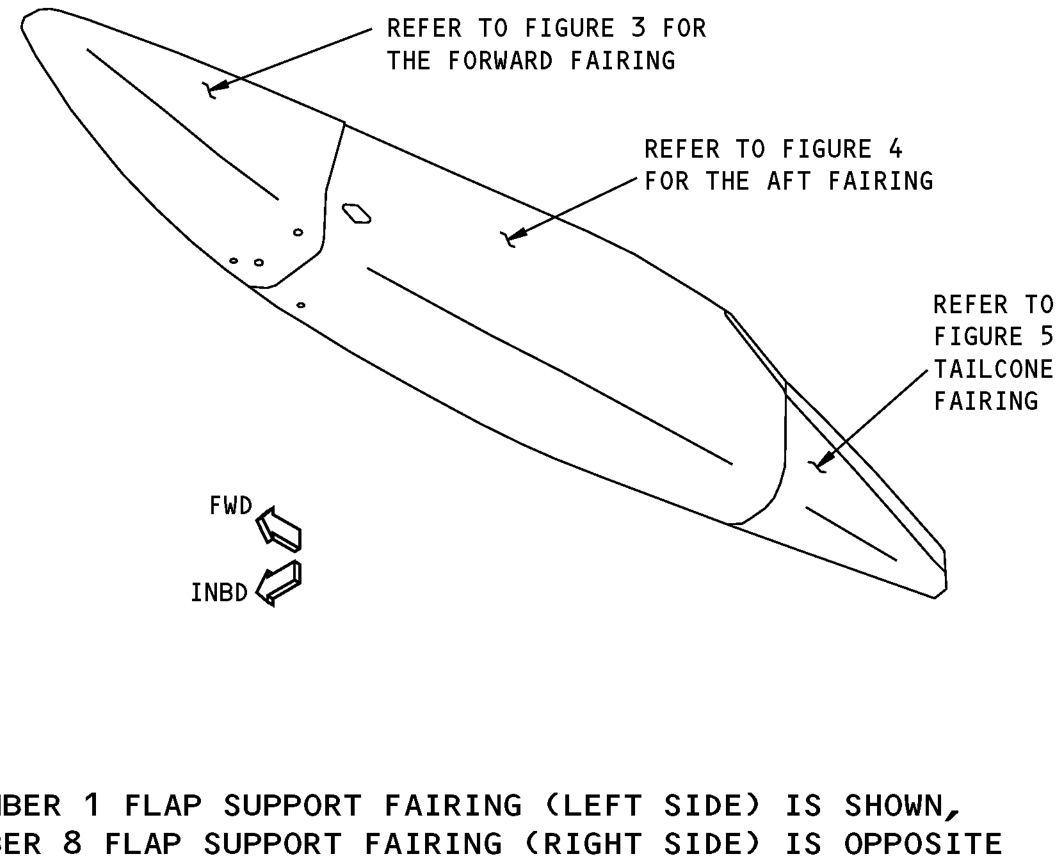
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Trailing Edge Flap Support Fairing Locations**
**Figure 1**

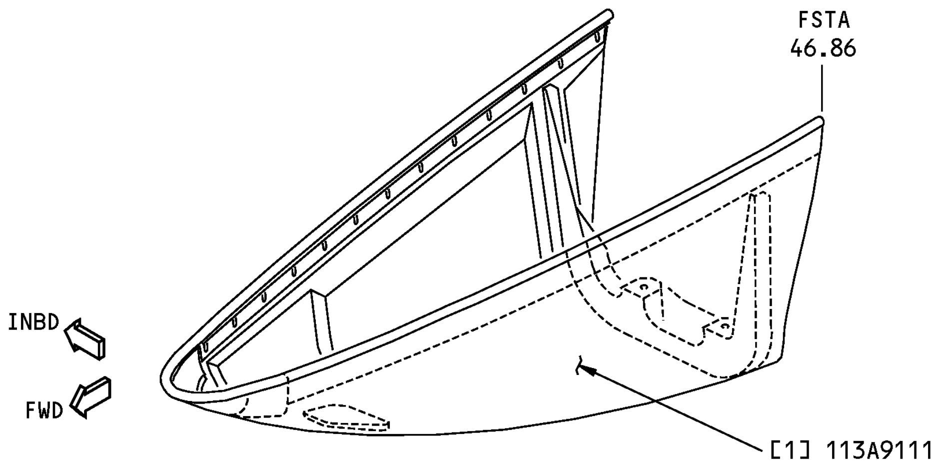
**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A9100 | Fairing Installation - Flap Support Numbers 1 and 8, Outboard Trailing Edge Flap |
| 113A9110 | Forward Fairing Assembly - Flap Support Number 1, Outboard Trailing Edge Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A9150 | Aft Fairing Assembly - Flap Support Number 1, Outboard Trailing Edge Flap |
| 113A9252 | Tailcone Assembly - Flap Support Fairing Numbers 1,2,7,8 |




**Outboard Trailing Edge Flap Support Fairing Skins**
**Figure 2**




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

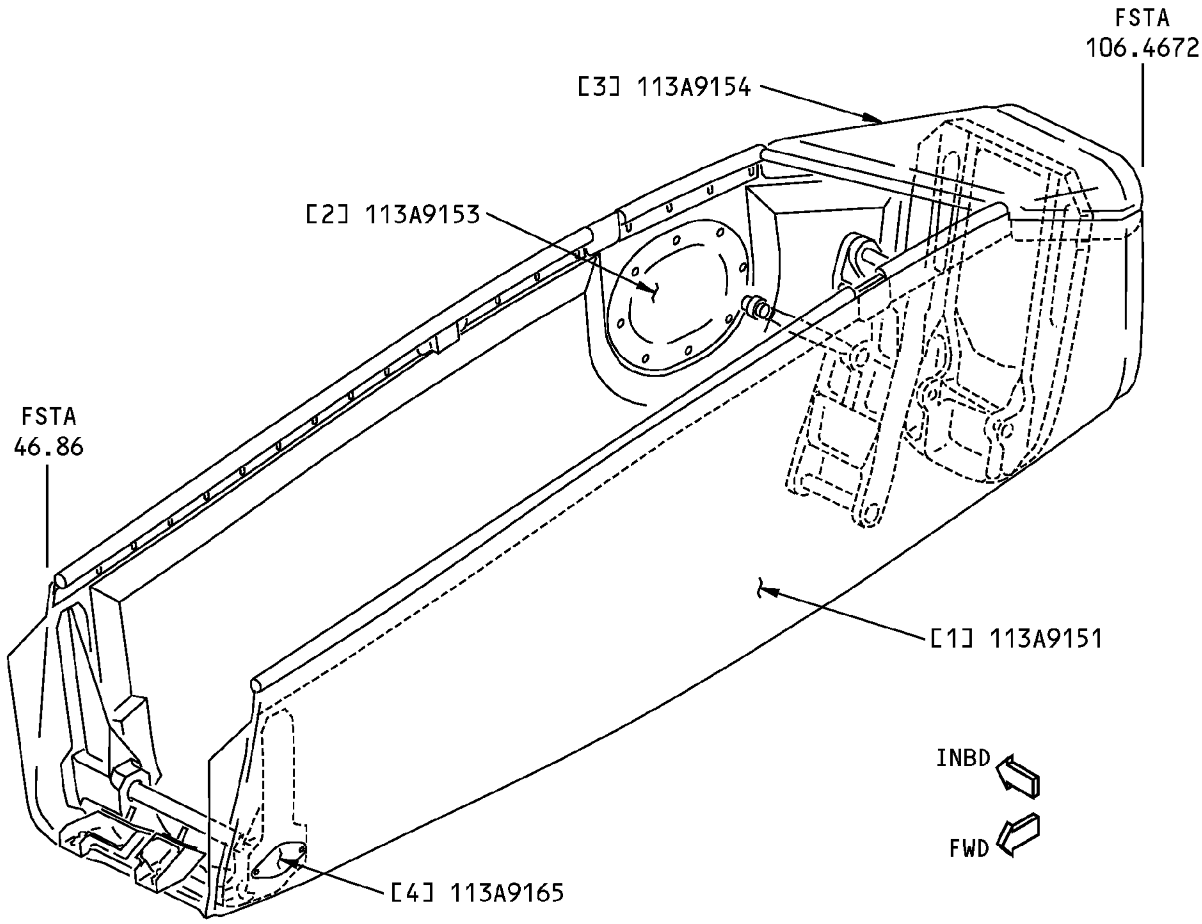
NUMBER 1 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN, NUMBER 8 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

**Outboard Trailing Edge Flap Support Forward Fairing Skin Identification**
**Figure 3**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Bonded Skin Assembly | | Glass Fiber Reinforced Plastic (GFRP) honeycomb sandwich | |
| | Assembly | | | |
| | Skin | | Refer to Figure 6 | |
| | Core (1) | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| | Core (1) | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

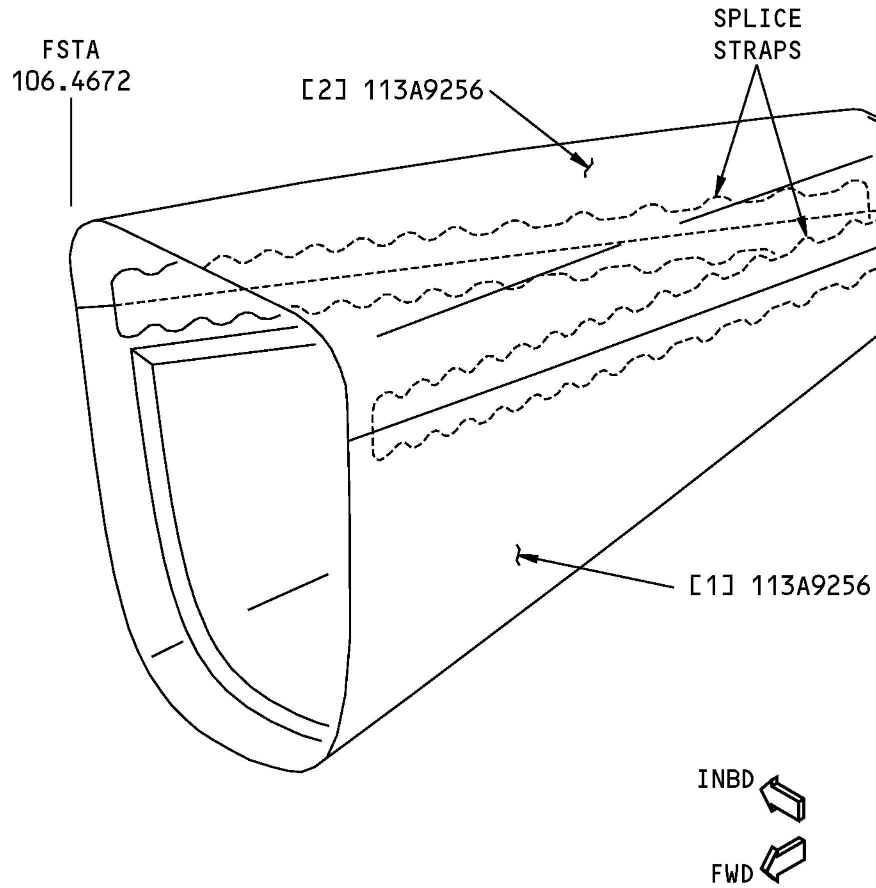
NUMBER 1 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 8 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

**Outboard Trailing Edge Flap Support Aft Fairing Skin Identification**
**Figure 4**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Bonded Skin Assembly | | GFRP and Carbon Fiber Reinforced Plastic (CFRP) honeycomb sandwich | |
| | Skin | | Refer to Figure 7 | |
| | Core | 0.50 (12.7) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [2] | Bonded Access Door Assembly | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) Refer to Figure 8 | |
| [3] | Bonded Aft Fairing Cap | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781, 120, or 220) Refer to Figure 9 | |
| [4] | Bonded Access Cover | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

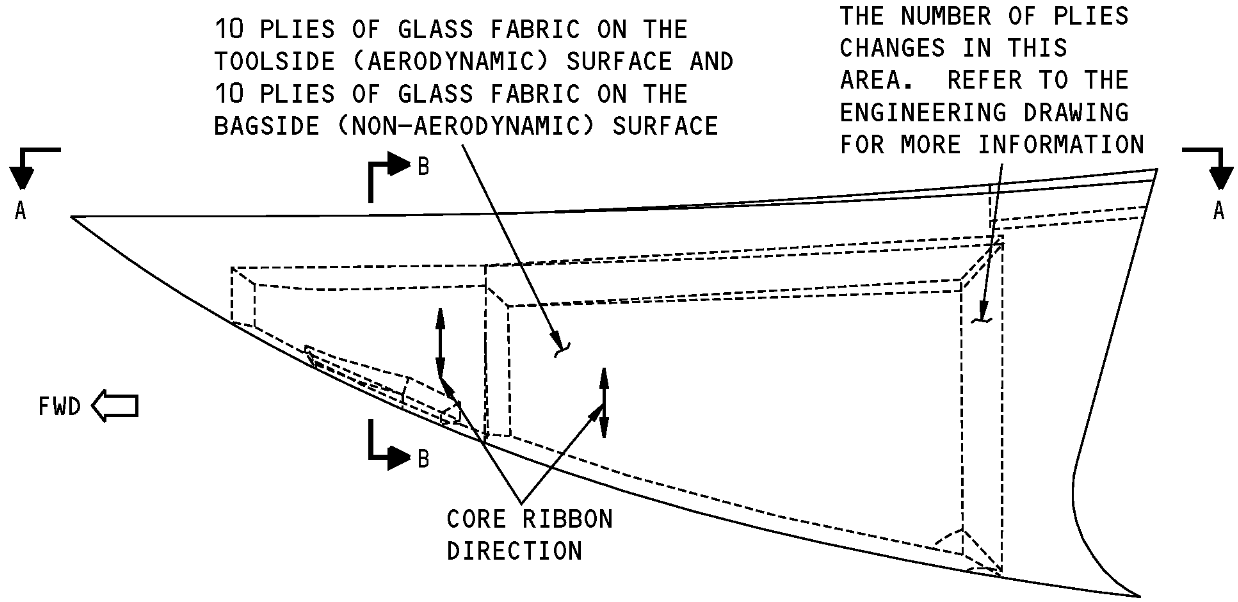
NUMBER 1 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 8 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

**Outboard Trailing Edge Flap Support Tailcone Fairing Skin Identification**
**Figure 5**

**Table 4**:

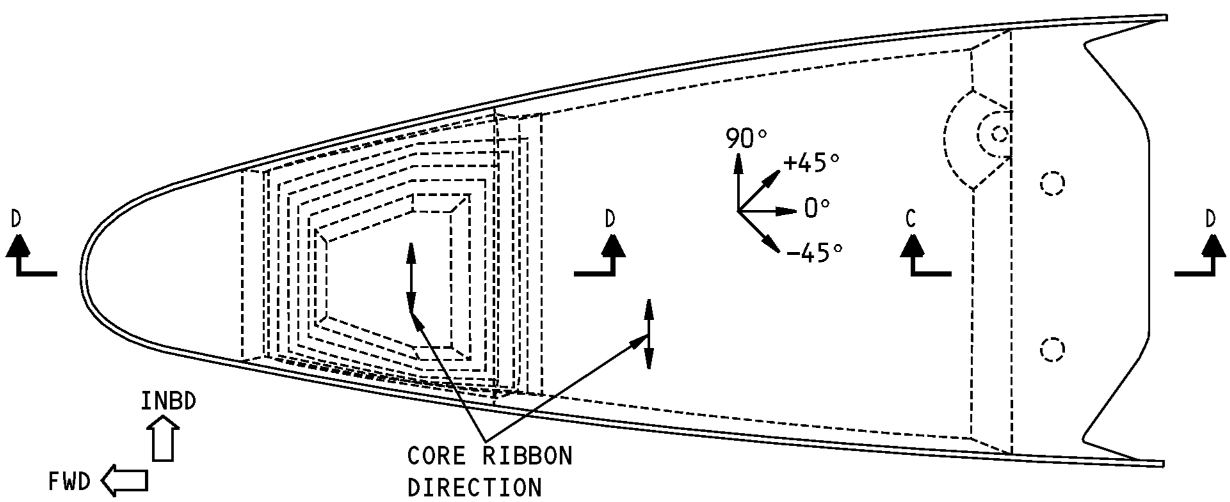
| LIST OF MATERIALS FOR FIGURE 5 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Lower Bonded Tailcone Assembly | | Glass epoxy honeycomb sandwich | |
| | Skin | | Refer to Figure 10 | |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [2] | Upper Bonded Tailcone Assembly | | Glass epoxy honeycomb sandwich | |
| | Skin | | Refer to Figure 11 | |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

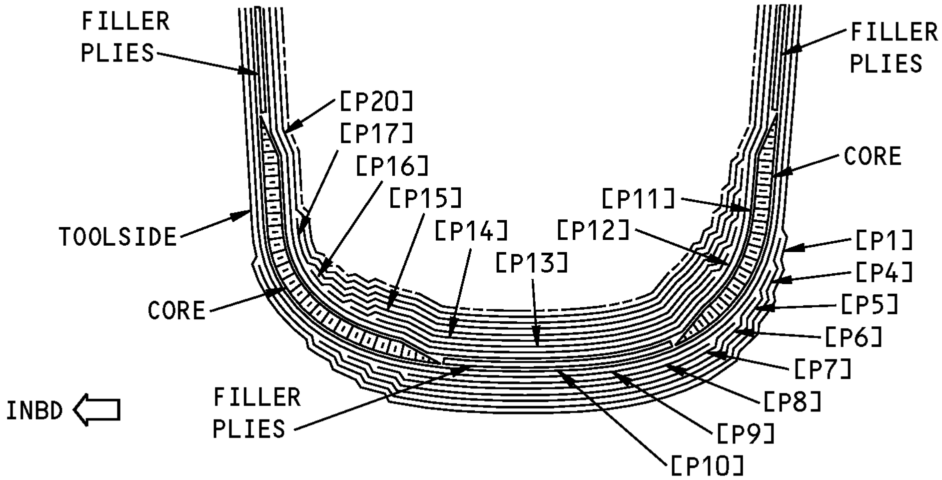



A-A

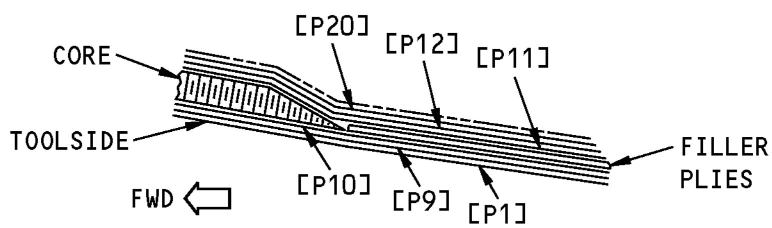
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B, C-C, AND D-D FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

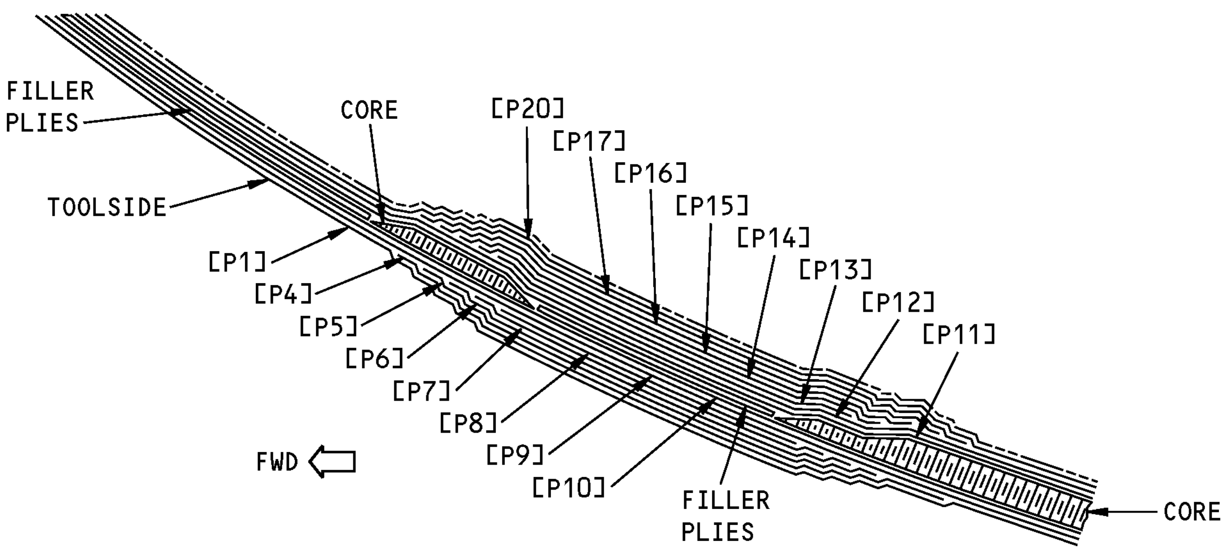
**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]**
**Figure 6 (Sheet 1 of 2)**




B-B




C-C



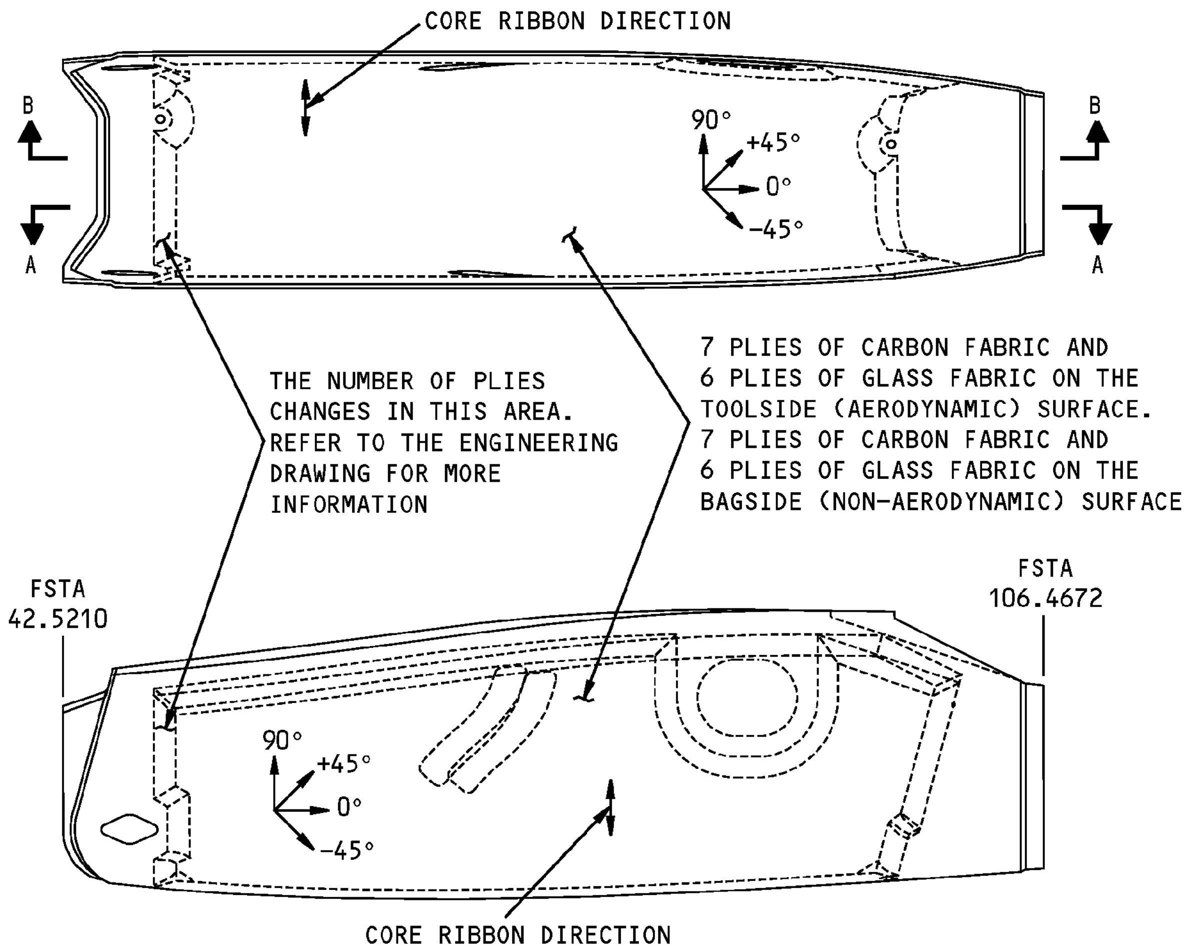

D-D

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]**
**Figure 6 (Sheet 2 of 2)**

**Table 5**:

| PLY MATERIAL AND DIRECTION OF THE FORWARD FAIRING BONDED SKIN FOR FIGURE 3 *[1] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P10, P11, and P20 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220) |
| P2 thru P4, P6, P8, P13, P15, and P17 thru P19 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| P5, P7, P9, P12, P14, and P16 | + or - 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220) |

*[1] Note: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.



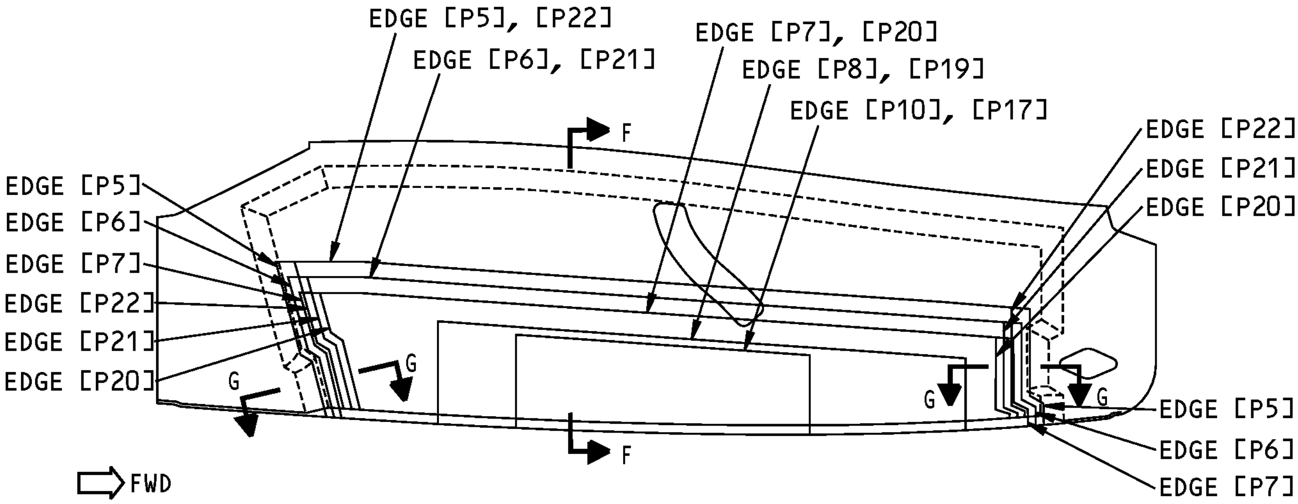

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

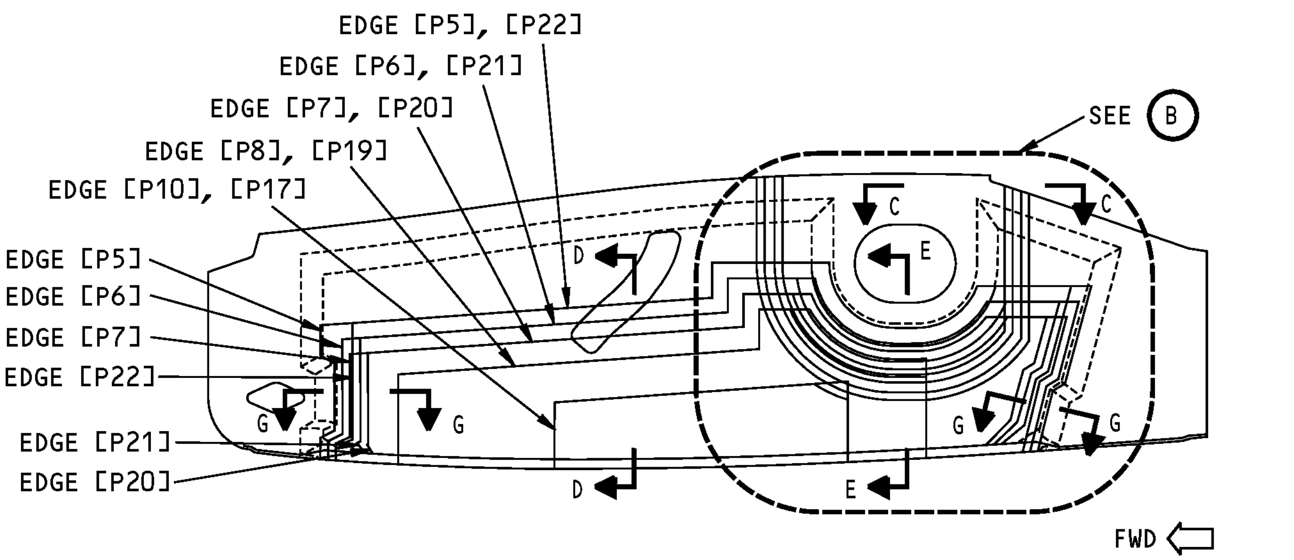
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, D-D, E-E, F-F, AND G-G FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 1 of 5)**




PLIES [P2], [P3], [P11], [P12], [P15], [P16], [P24], AND [P25] NOT SHOWN FOR CLARITY

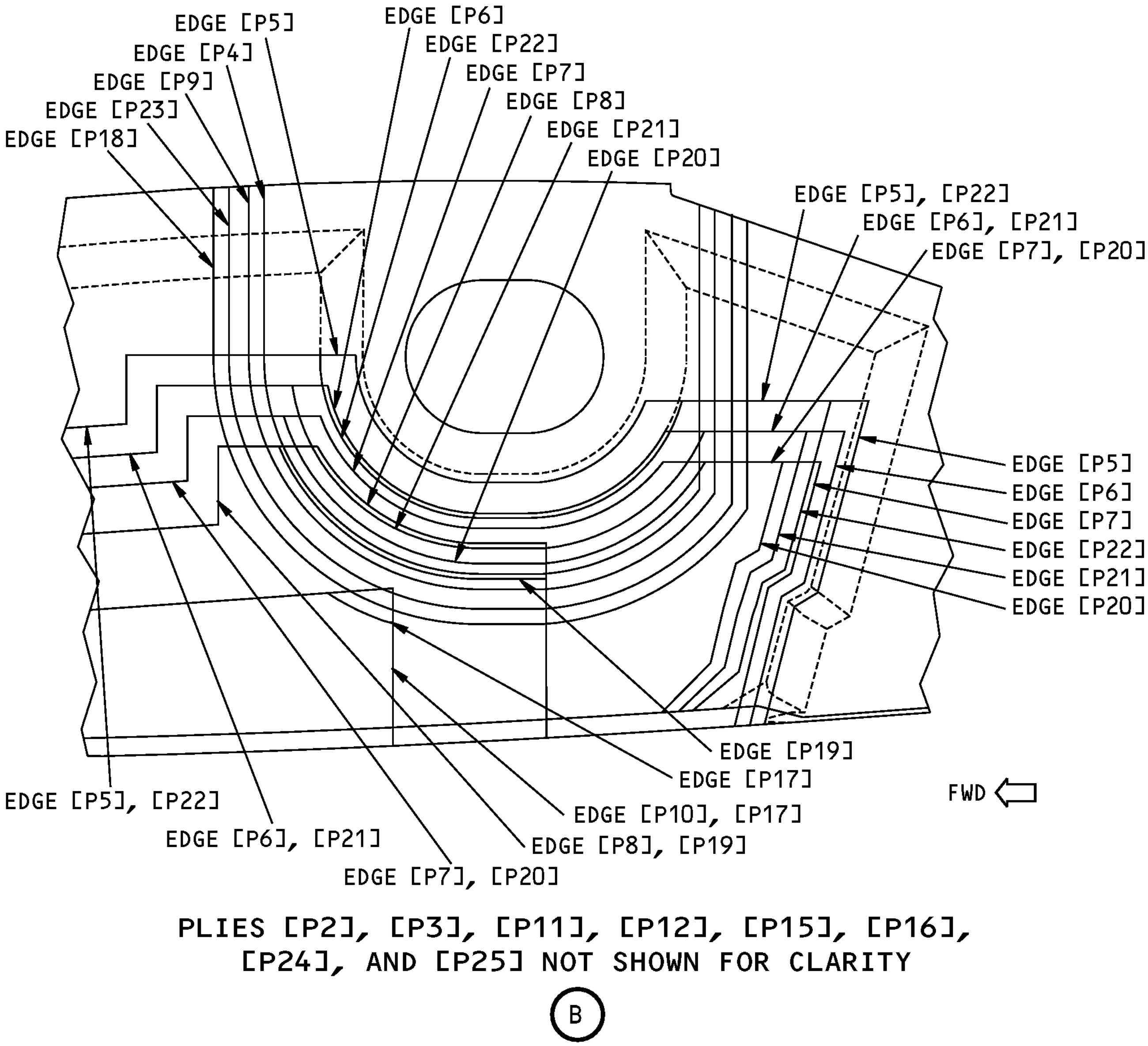
A-A

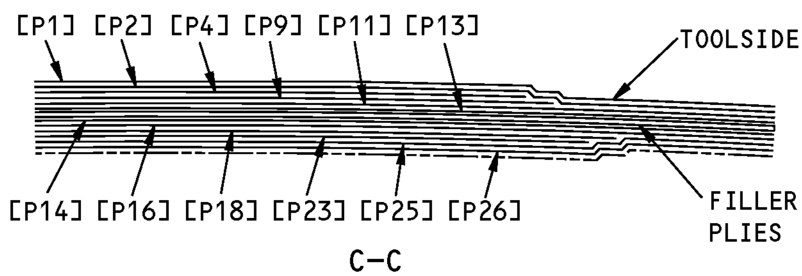



PLIES [P2], [P3], [P11], [P12], [P15], [P16], [P24], AND [P25] NOT SHOWN FOR CLARITY

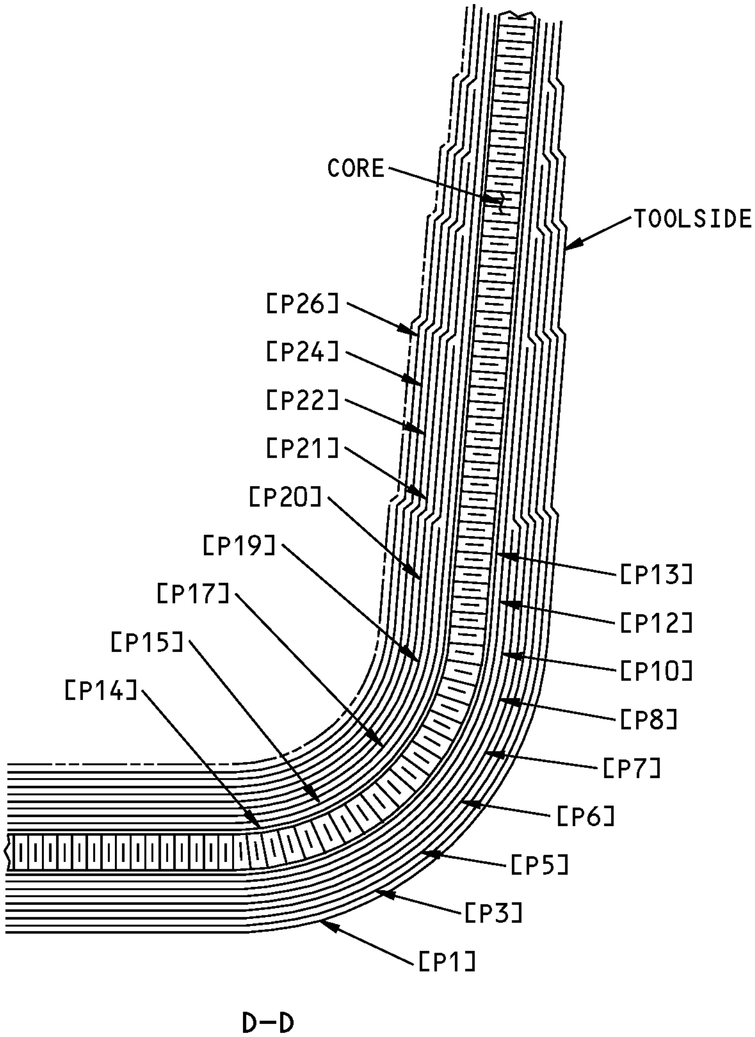
B-B

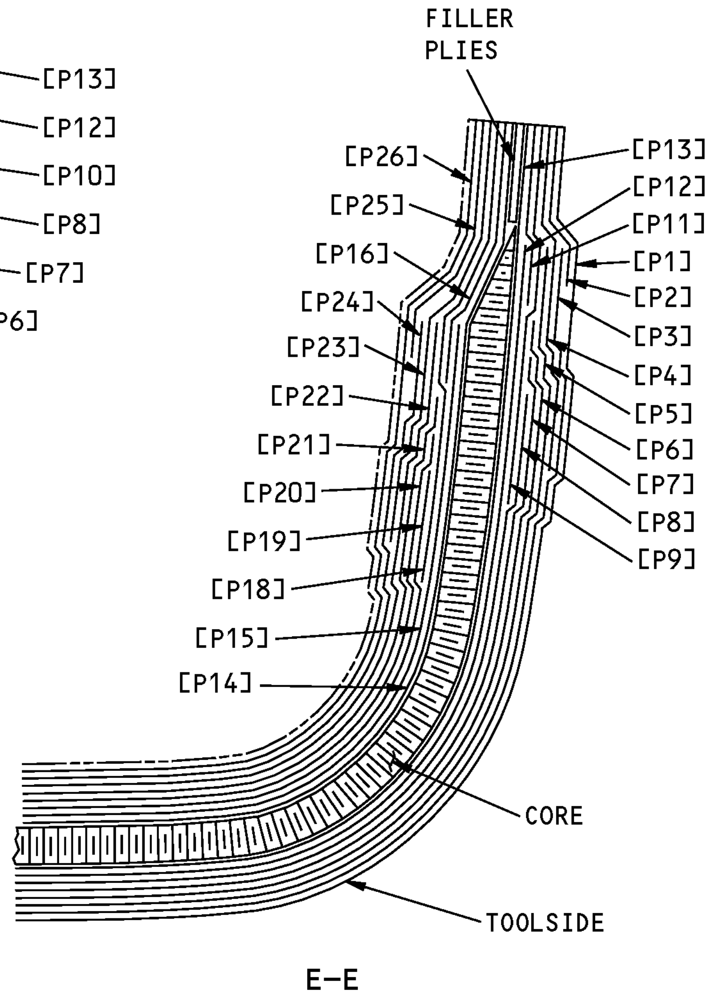
**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 2 of 5)**



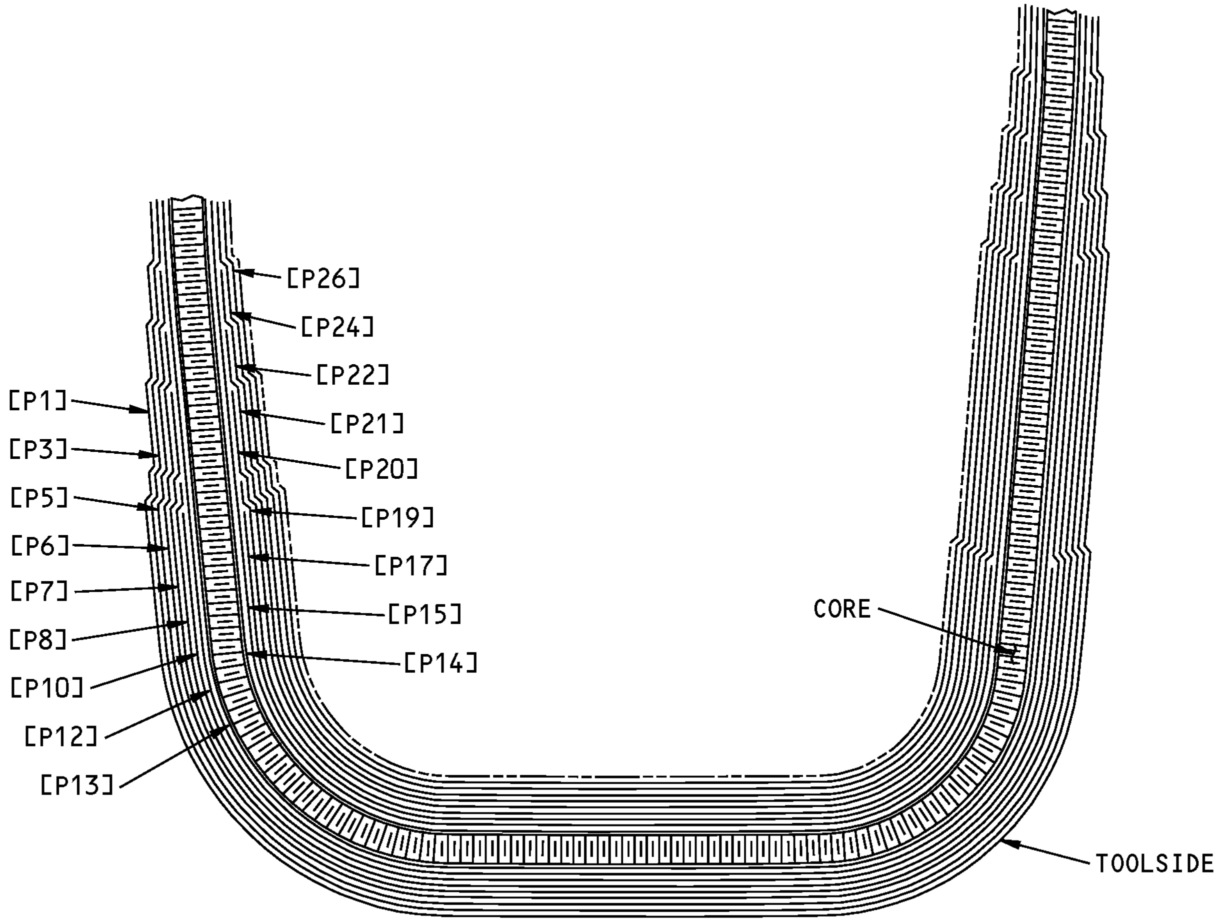




**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 3 of 5)**

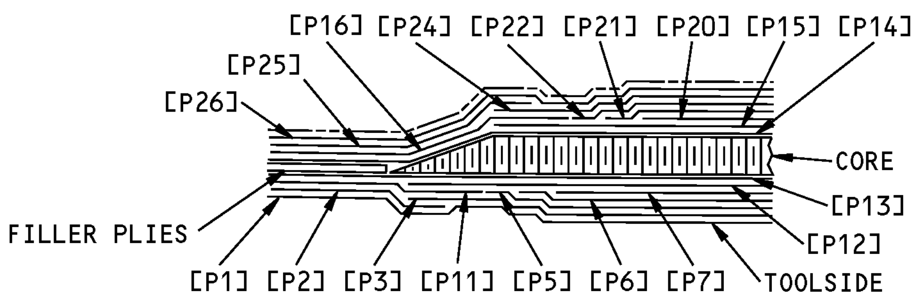



**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 4 of 5)**




F-F



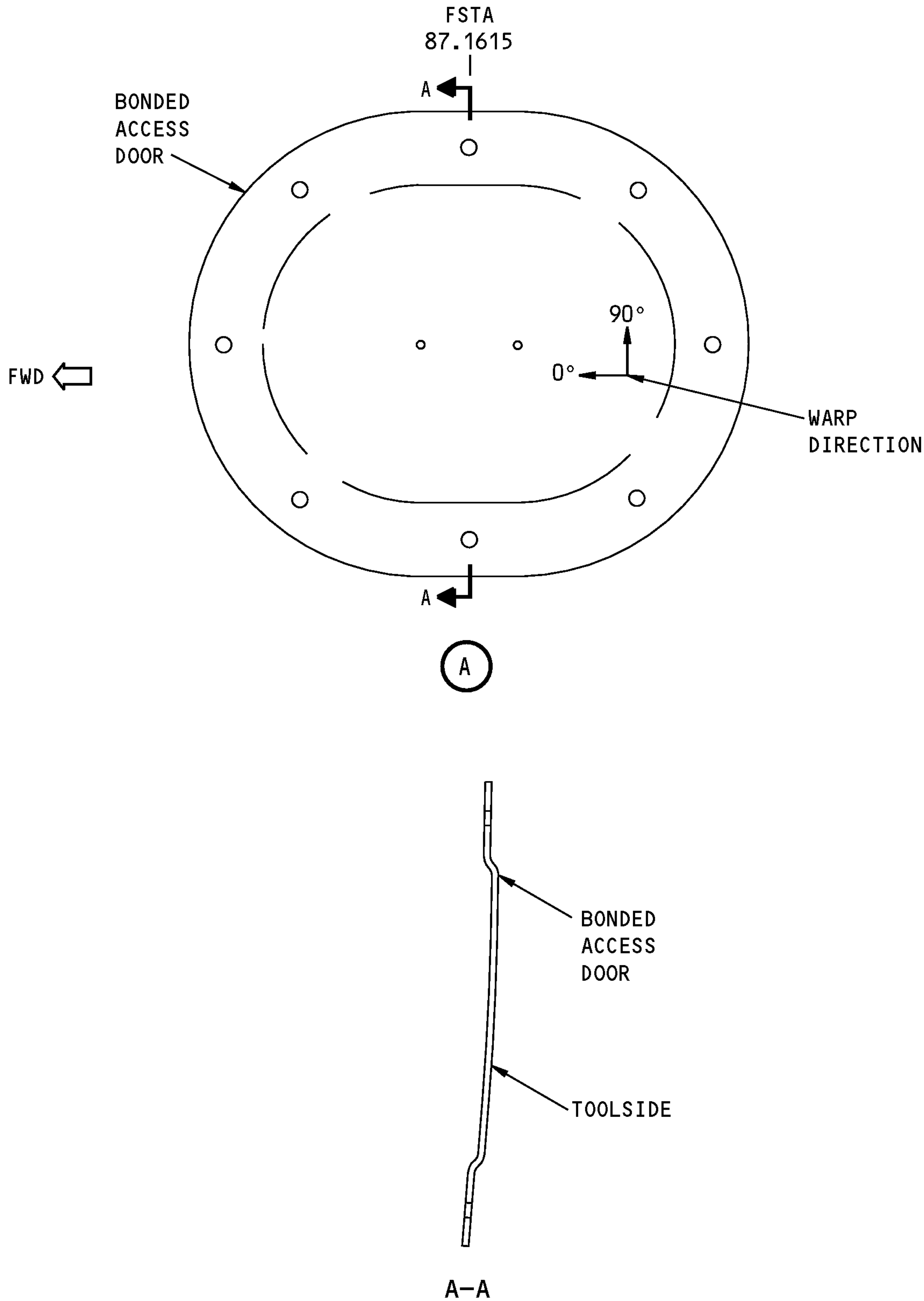

G-G

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 5 of 5)**

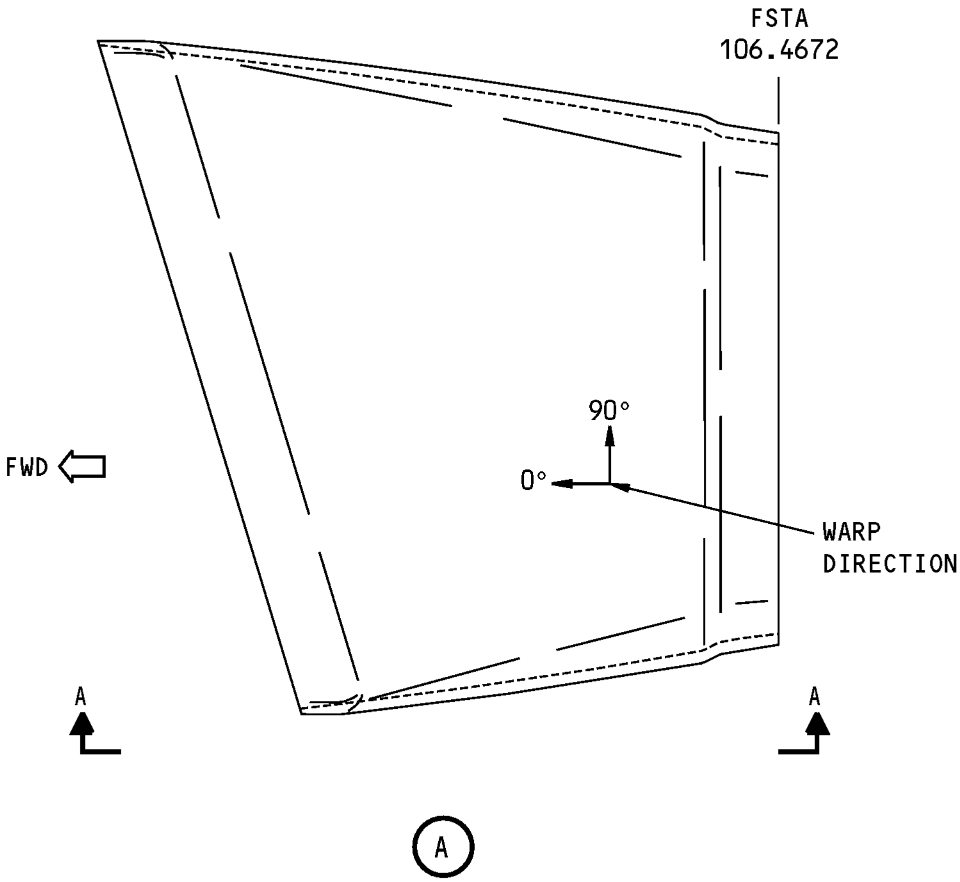
**Table 6**:

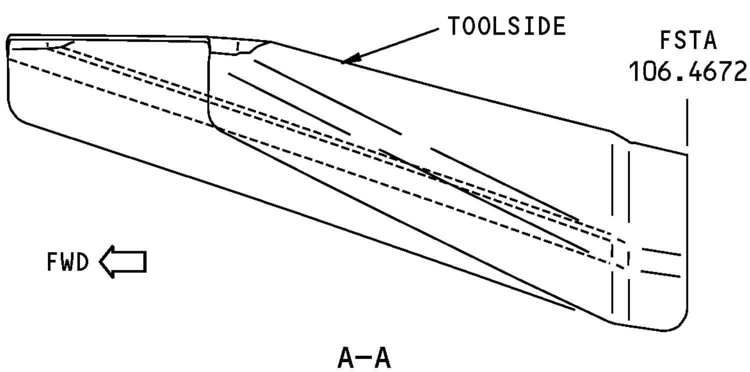
| PLY MATERIAL AND DIRECTION FOR FIGURE 7*[1] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P13, P14, and P26 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220) |
| P2, P11, P16, and P25 | + or - 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| P3, P12, P15, and P24 | + or - 45 degrees | CFRP as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW |
| P4, P9, P18, and P23 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| P5 thru P8, P10, P17, and P19 thru P22 | 0 or 90 degrees | CFRP as given in BMS 8-168, Type I, Class 2, Style 3K-70-PW |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220) |

*[1] Note: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.

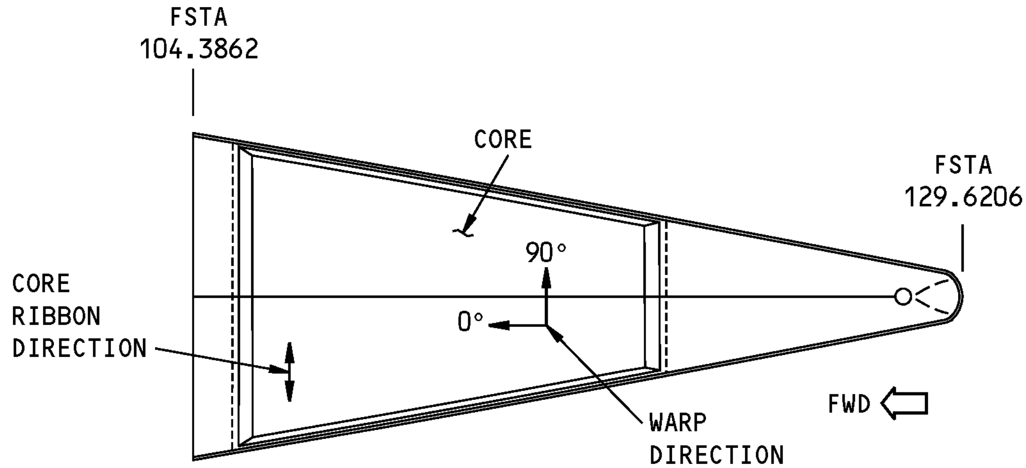


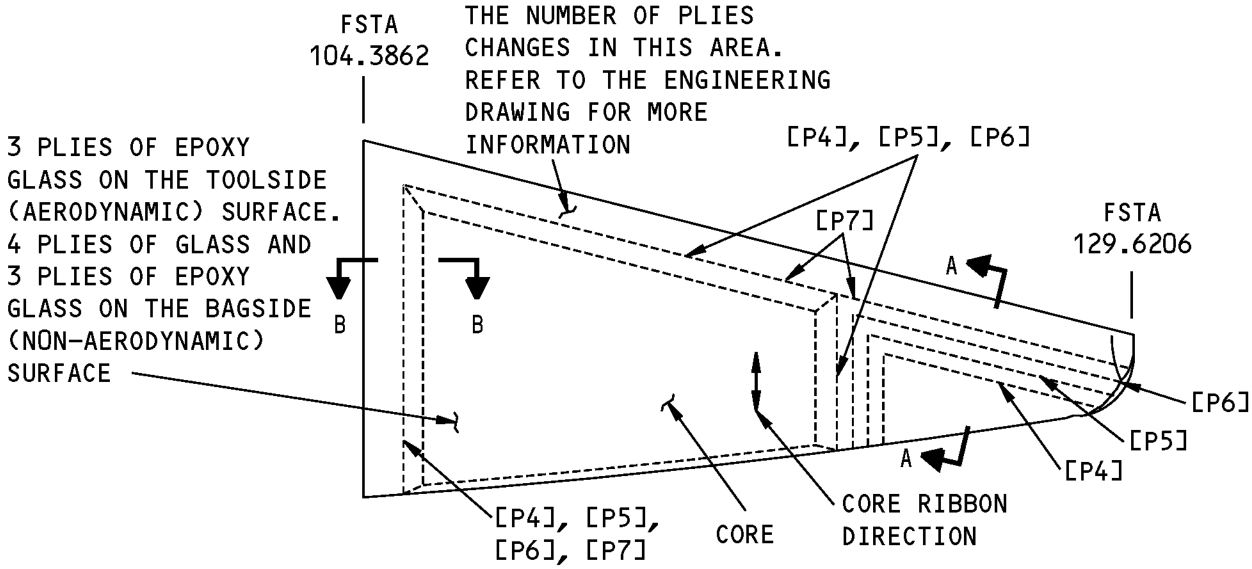

**Warp Direction of Material for Bonded Access Door, Figure 4, Item [2]**
**Figure 8**




**Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [3]**
**Figure 9**



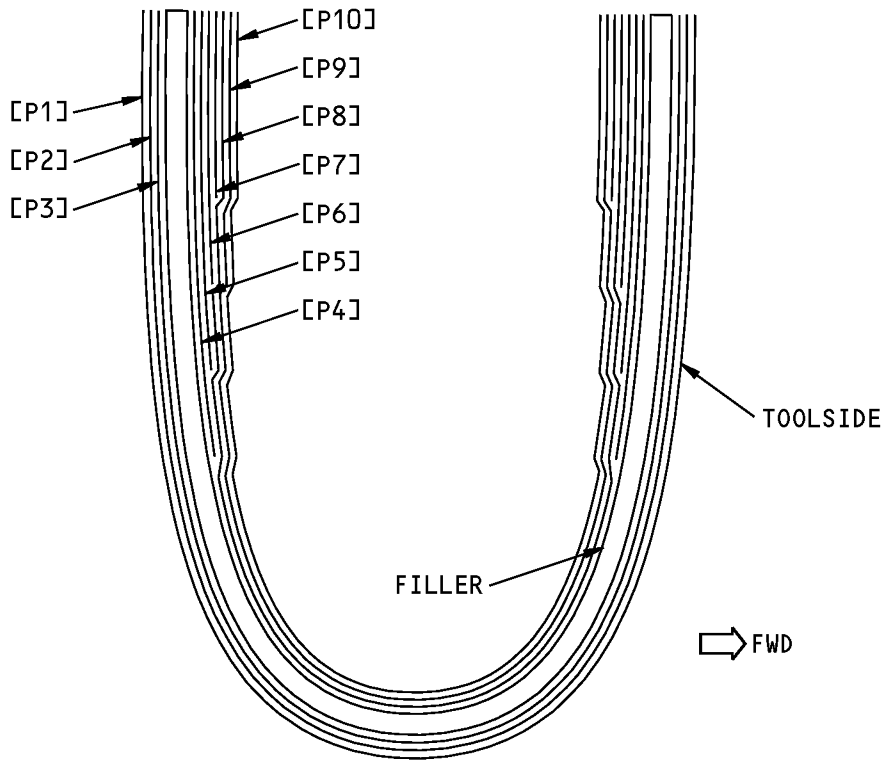

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

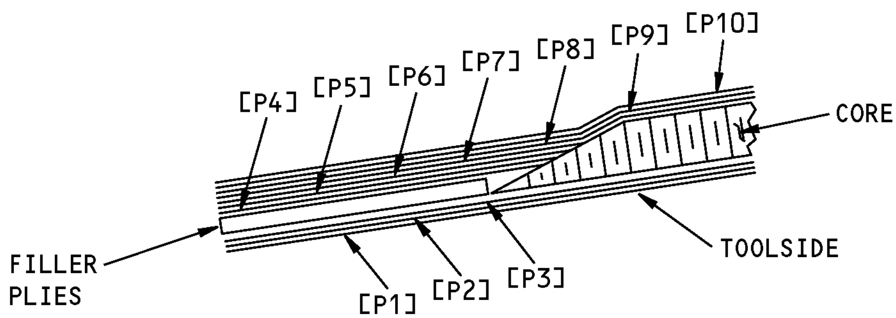
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 10 (Sheet 1 of 2)**




A-A

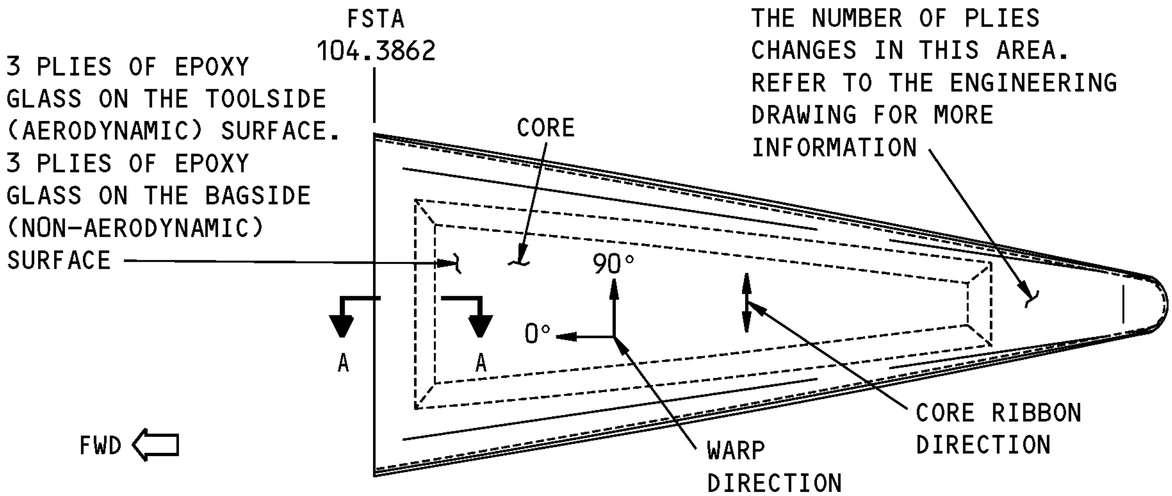



B-B

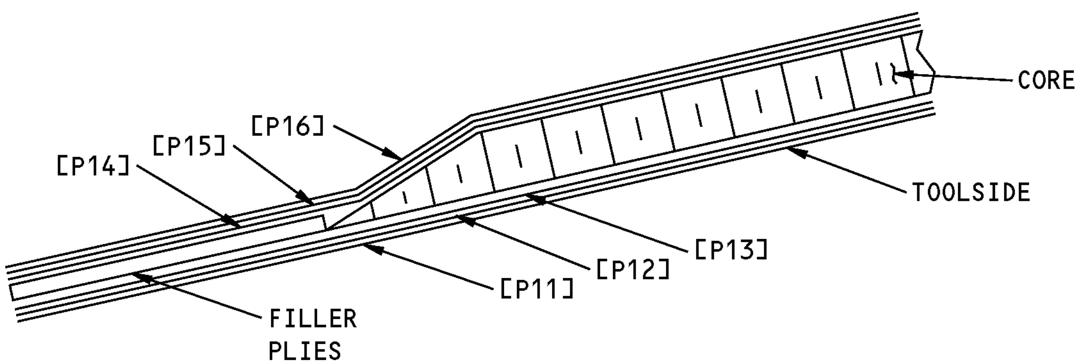
**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 10 (Sheet 2 of 2)**

**Table 7**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 10 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 thru P3, and P8 thru P10 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220) |
| P4 thru P7 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781) |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION
A




A-A

NOTES

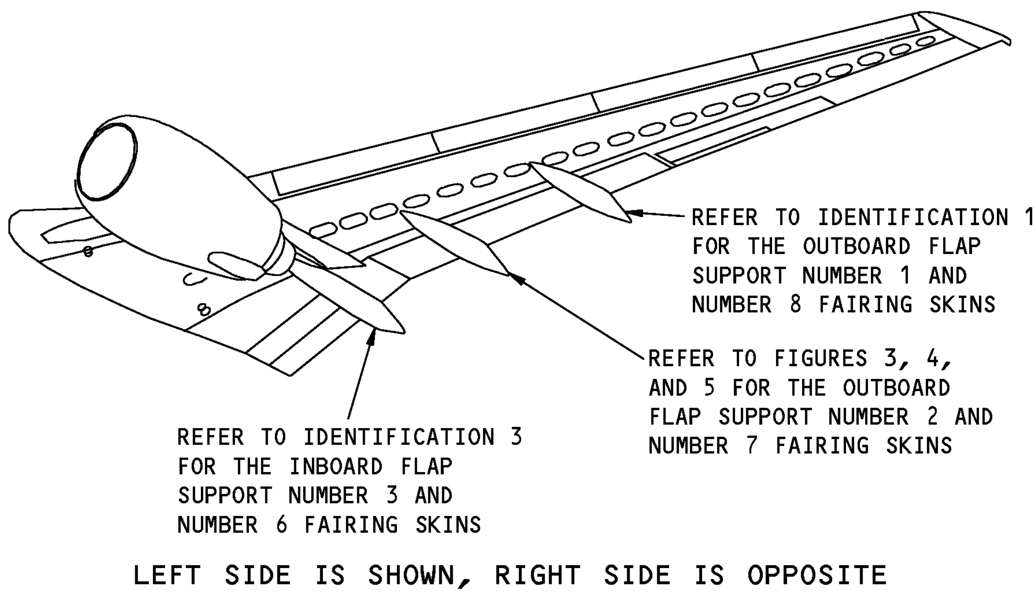
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]**
**Figure 11**

**Table 8**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 11 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P11 thru P16 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781) |

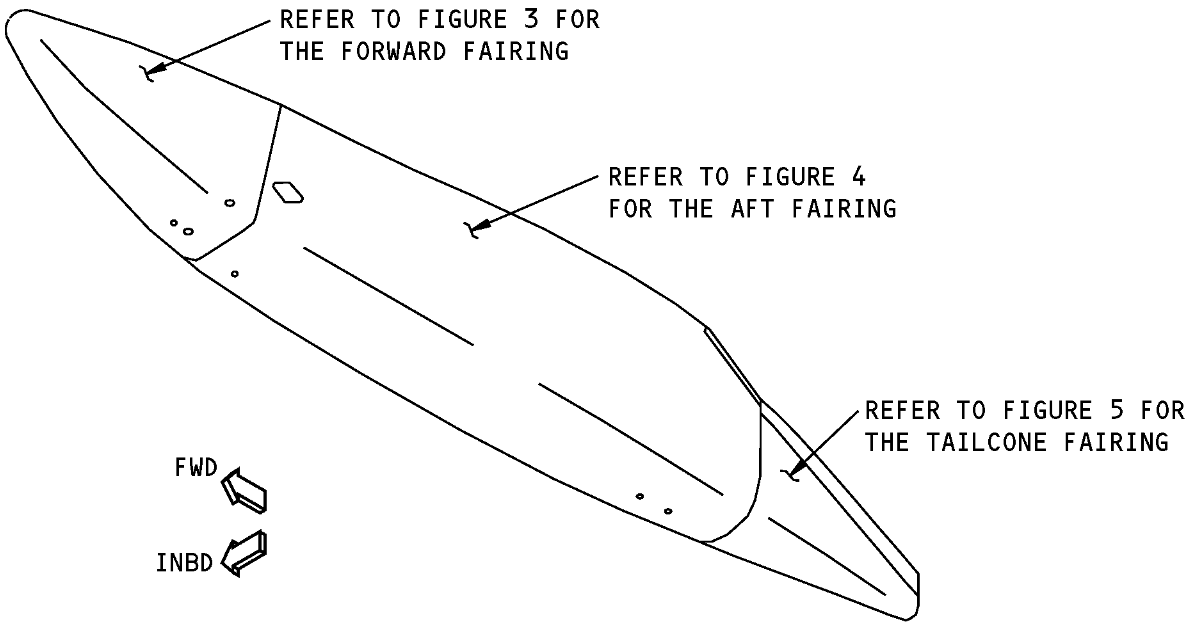
## IDENTIFICATION 2 - FAIRING SKINS - WING OUTBOARD TRAILING EDGE FLAP SUPPORTS - NUMBERS 2 AND 7




**Wing Trailing Edge Flap Support Fairing Locations**
**Figure 1**

**Table 1**:

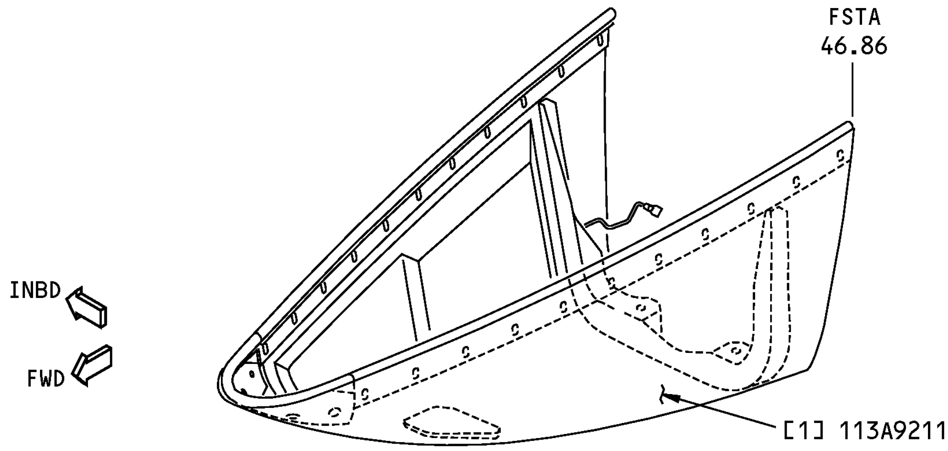
| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A9200 | Fairing Installation - Flap Support Numbers 2 and 7, Outboard Trailing Edge Flap |
| 113A9210 | Forward Fairing Assembly - Flap Support Number 2, Outboard Trailing Edge Flap |
| 113A9250 | Aft Fairing Assembly - Flap Support Number 2, Outboard Trailing Edge Flap |
| 113A9252 | Tailcone Assembly - Flap Support Fairing Numbers 1,2,7,8 |




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

NUMBER 2 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

**Outboard Trailing Edge Flap Support Fairing Skins**
**Figure 2**




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

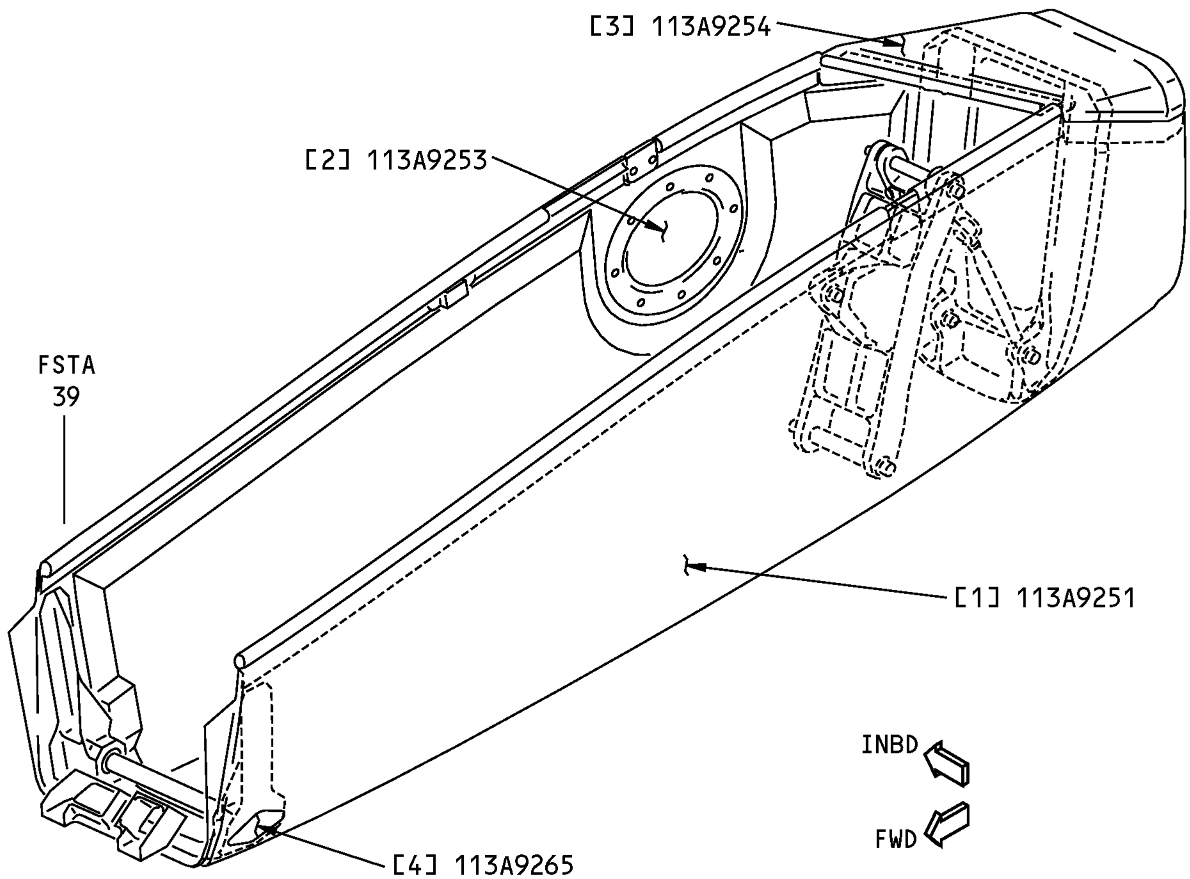
NUMBER 2 FLAP SUPPORT FORWARD FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT FORWARD FAIRING (RIGHT SIDE) IS OPPOSITE

**Outboard Trailing Edge Flap Support Forward Fairing Skin Identification**
**Figure 3**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Bonded Skin Assembly | | Glass Fiber Reinforced Plastic (GFRP) honeycomb sandwich | |
| | Skin | | Refer to Figure 6 | |
| | Core (2) | | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

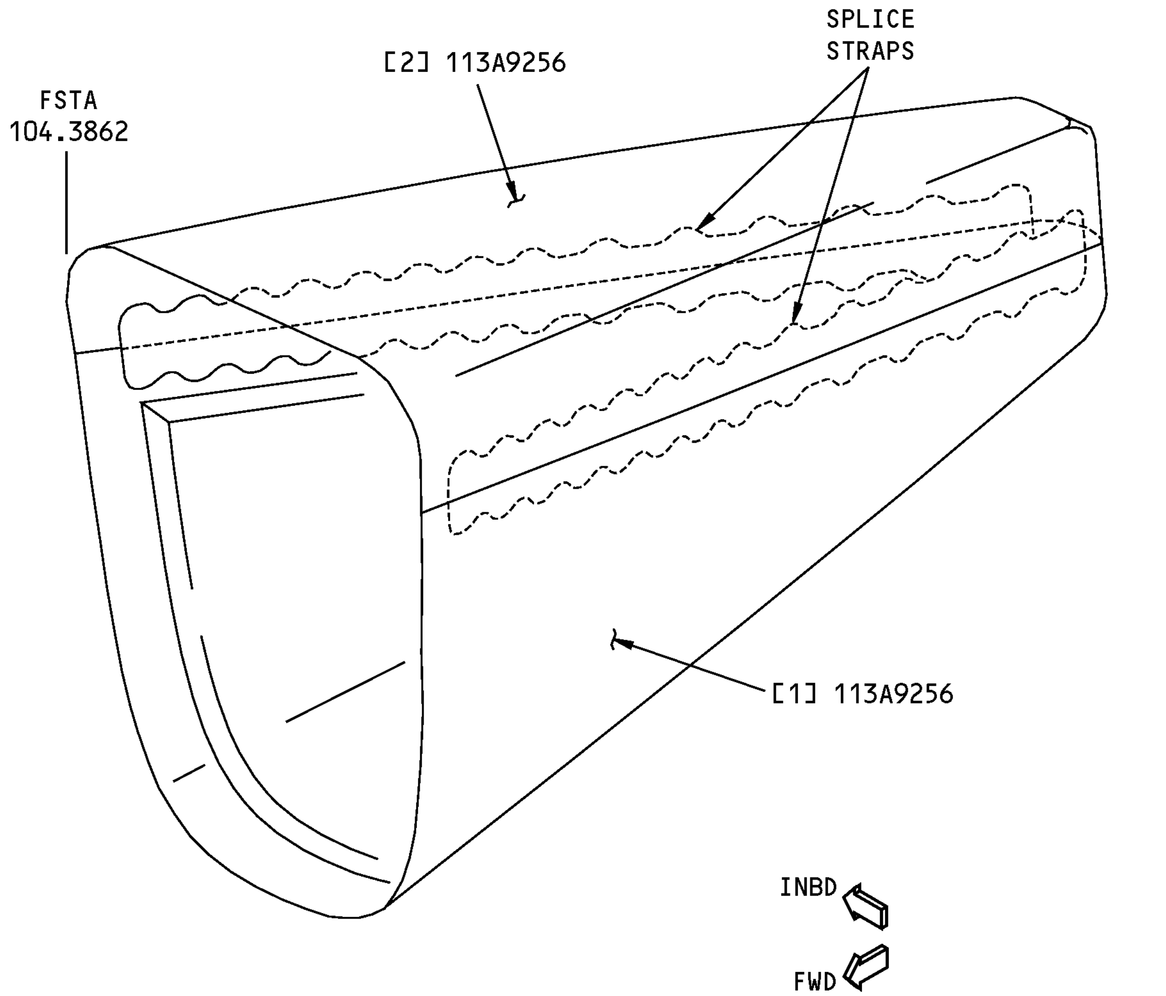
NUMBER 2 FLAP SUPPORT AFT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT AFT FAIRING (RIGHT SIDE) IS OPPOSITE

**Outboard Trailing Edge Flap Support Aft Fairing Skin Identification**
**Figure 4**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Bonded Skin Assembly | | CFRP and GFRP honeycomb sandwich | |
| | Skin | | Refer to Figure 7 | |
| | Core (2) | 0.50 (12.7)<br>0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [2] | Bonded Access Door Assembly | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781). Refer to Figure 8 | |
| [3] | Bonded Aft Fairing Cap | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781, 120, or 220). Refer to Figure 9 | |
| [4] | Bonded Access Cover | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

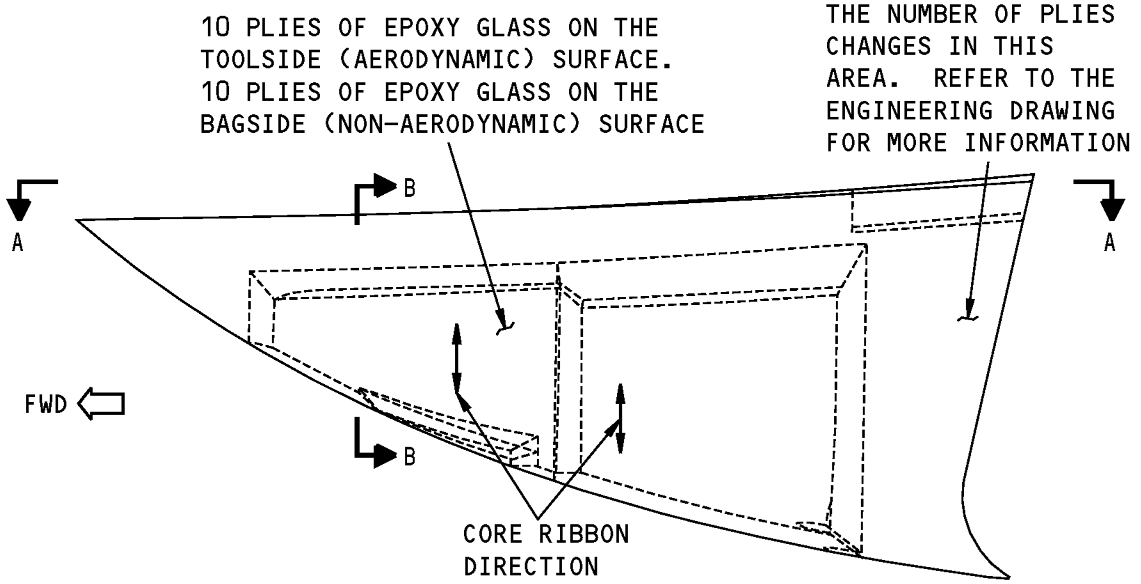
NUMBER 2 FLAP SUPPORT TAILCONE FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT TAILCONE FAIRING (RIGHT SIDE) IS OPPOSITE

**Outboard Trailing Edge Flap Support Tailcone Fairing Skin Identification**
**Figure 5**

**Table 4**:

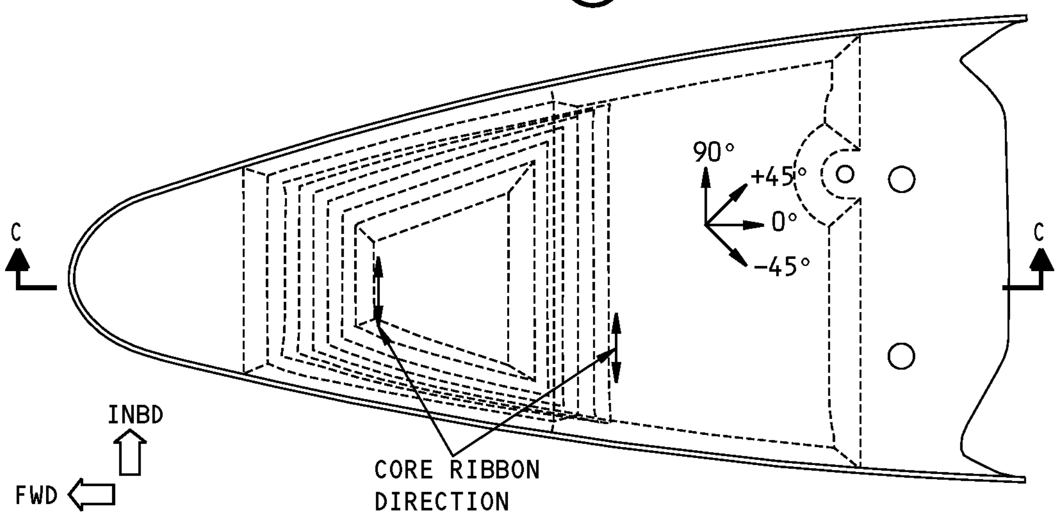
| LIST OF MATERIALS FOR FIGURE 5 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Lower Bonded Tailcone Assembly | | GFRP honeycomb sandwich | |
| | Skin | | Refer to Figure 10 | |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [2] | Upper Bonded Tailcone Assembly | | GFRP honeycomb sandwich | |
| | Skin | | Refer to Figure 11 | |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

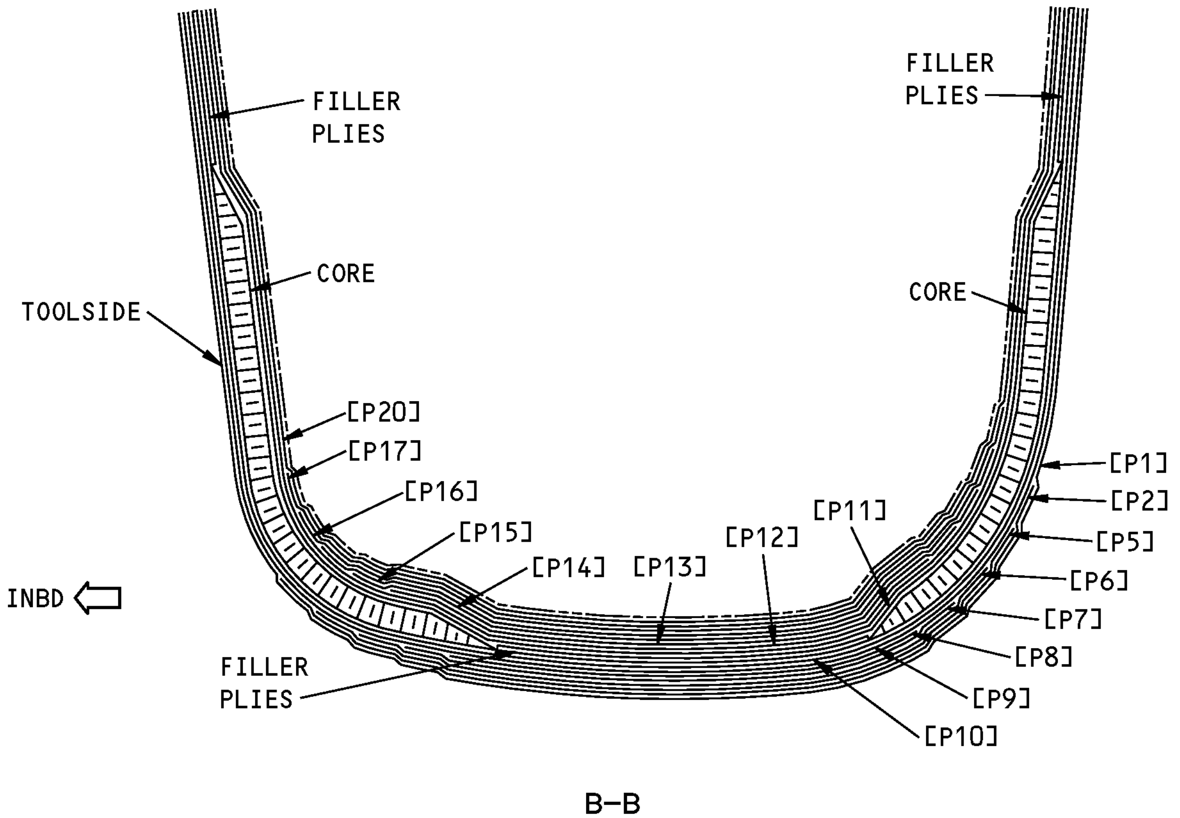


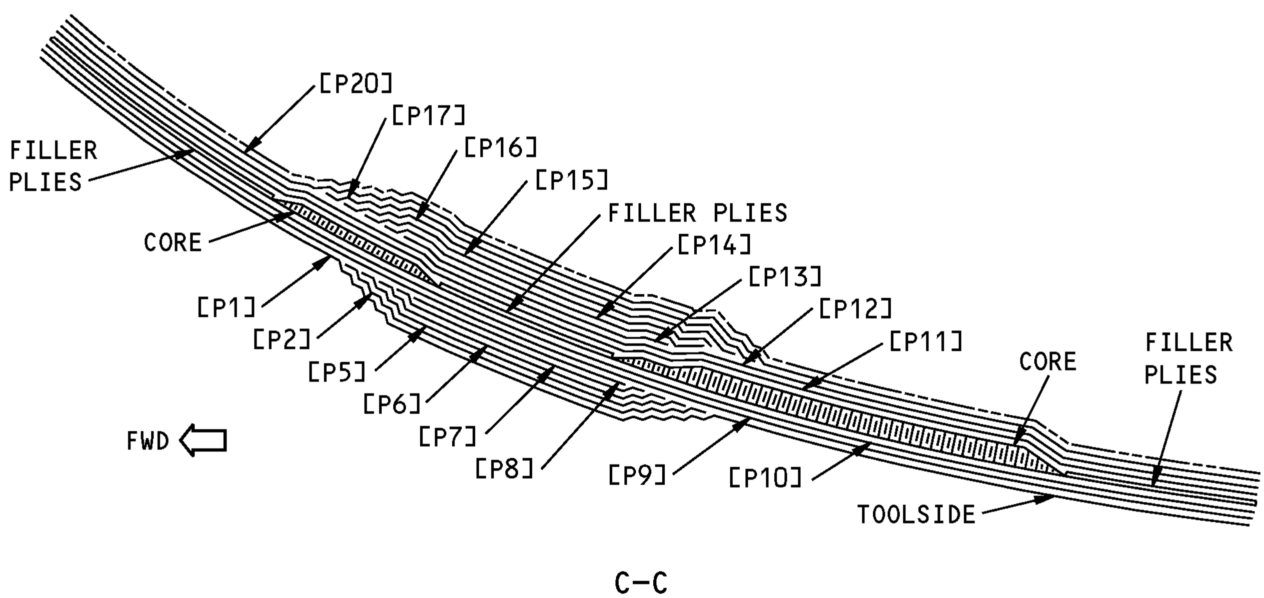

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO SECTION A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B AND C-C FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]**
**Figure 6 (Sheet 1 of 2)**



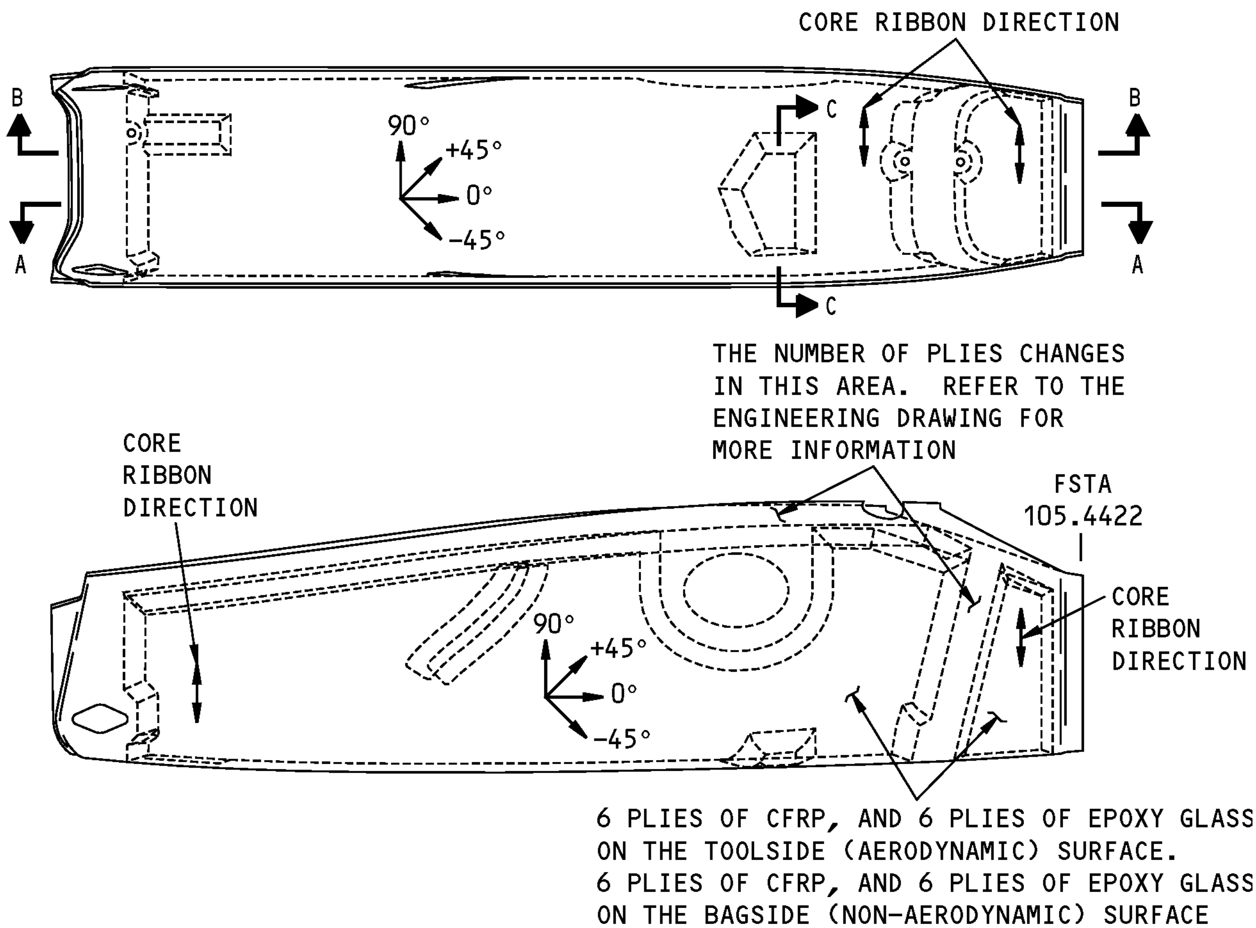




**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]**
**Figure 6 (Sheet 2 of 2)**

**Table 5**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 6 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P10, P11, and P20 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120. (Optional: Style 220) |
| P2 thru P4, P6, P8, P13, P15, and P17 thru P19 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| P5, P7, P9, P12, P14, and P16 | + or - 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220) |

**NOTE**: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.



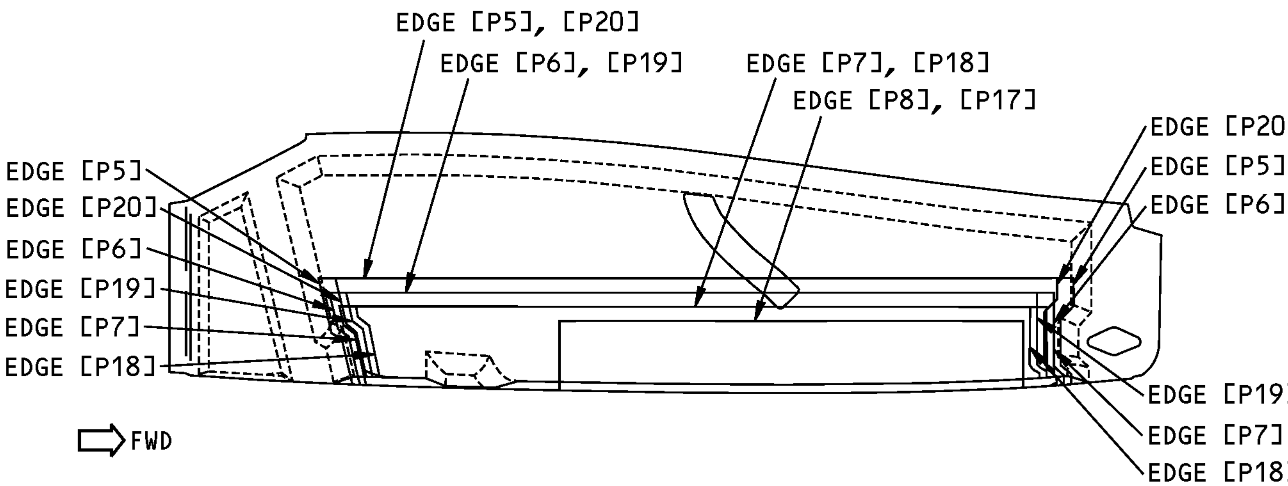

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

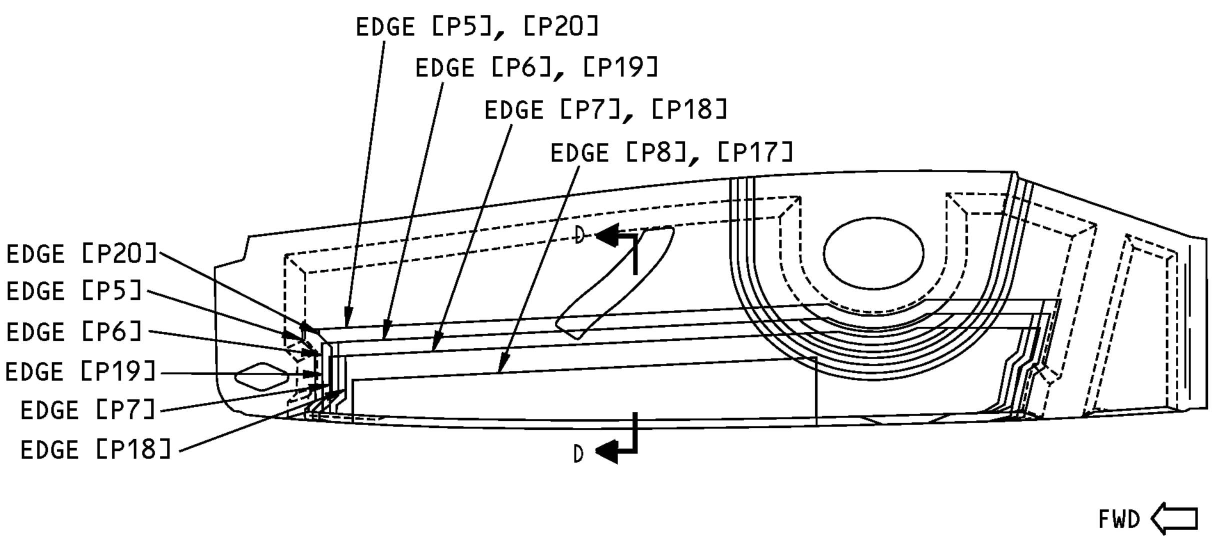
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, AND D-D FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 1 of 3)**




PLIES [P2], [P3], [P10], [P11], [P14], [P15],
[P22], AND [P23] NOT SHOWN FOR CLARITY

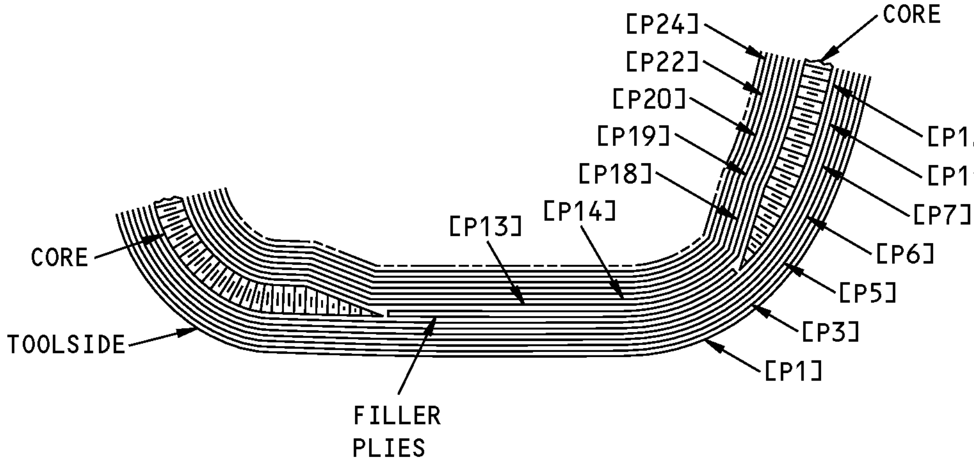
A-A

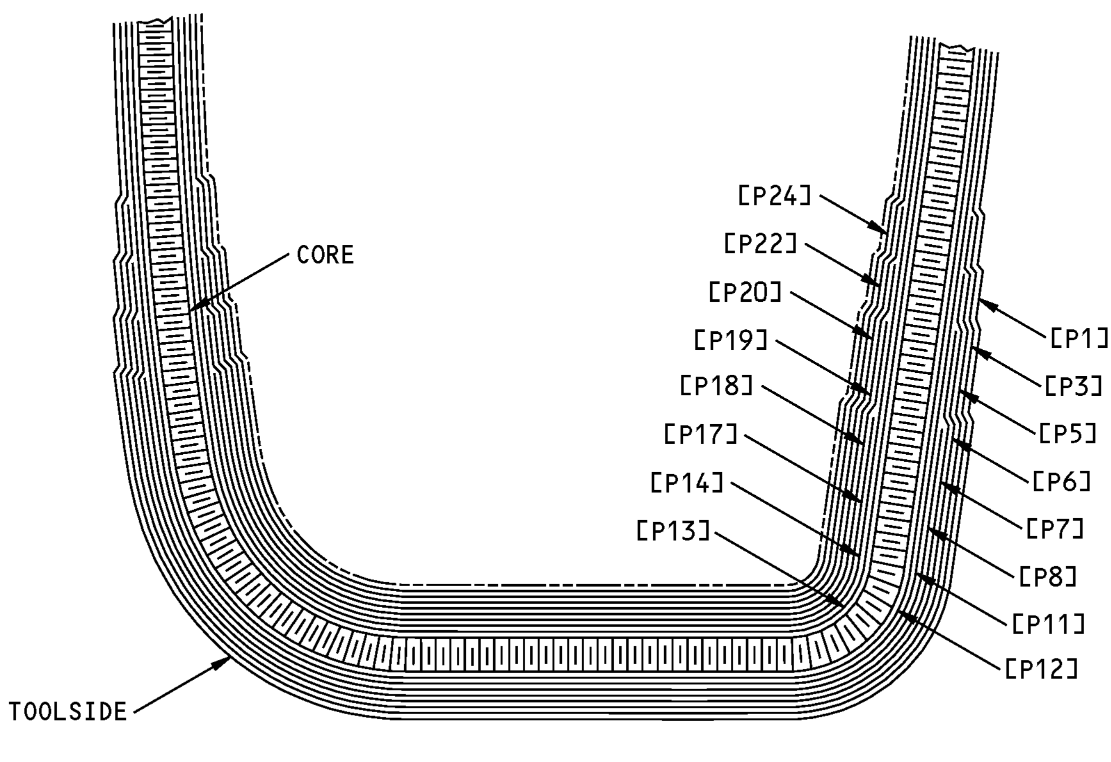



PLIES [P2], [P3], [P10], [P11], [P14], [P15],
[P22], AND [P23] NOT SHOWN FOR CLARITY

B-B

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 2 of 3)**



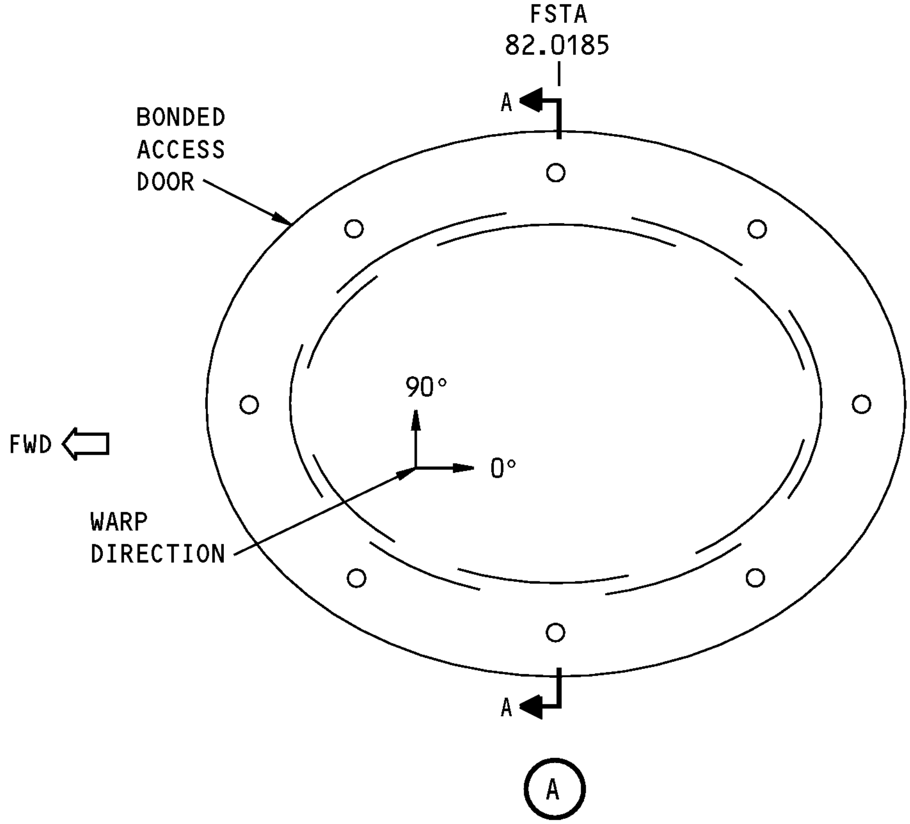



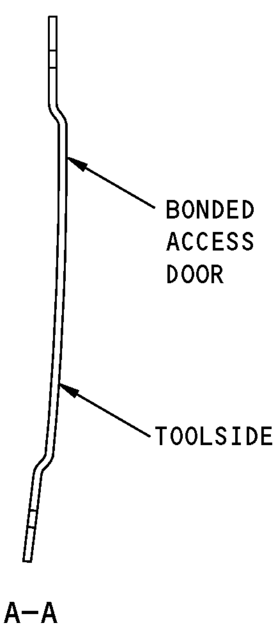

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 3 of 3)**

**Table 6**:

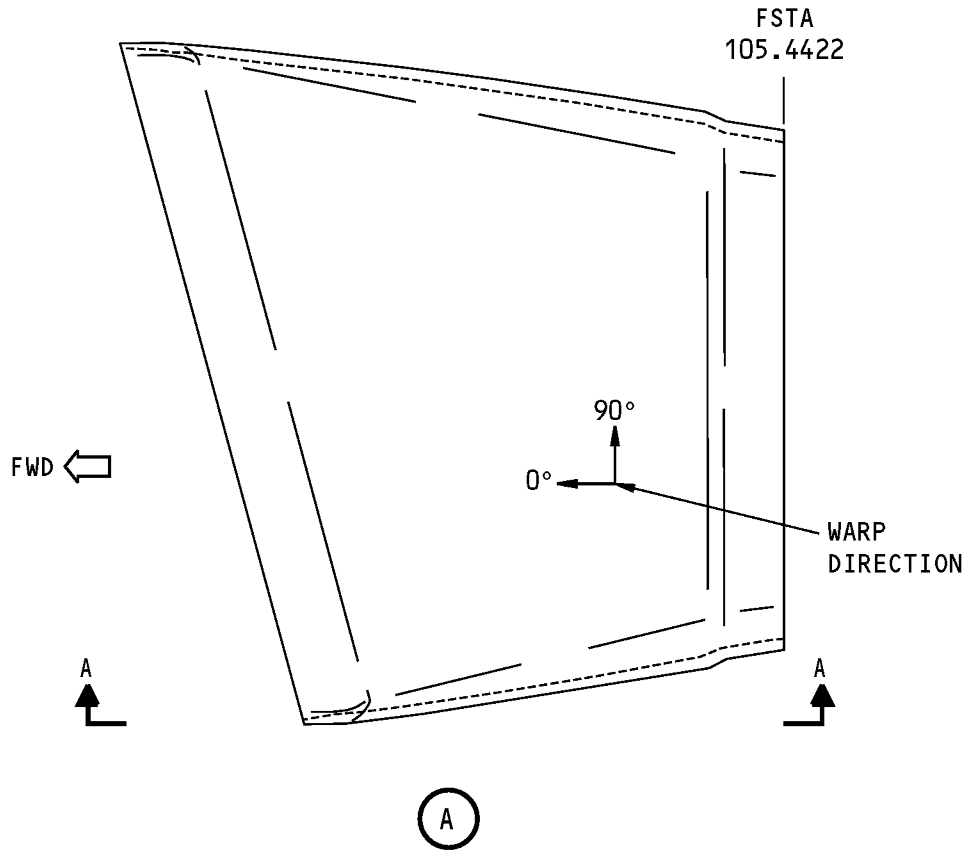
| PLY MATERIAL AND DIRECTION FOR FIGURE 7 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P12, P13, and P24 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120. (Optional: Style 220) |
| P2, P10, P15, and P23 | + or - 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| P3, P11, P14, and P22 | + or - 45 degrees | Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW |
| P4, P9, P16, and P21 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| P5 thru P8, and P17 thru P20 | 0 or 90 degrees | Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW |
| Filler Plies | + or - 45 degrees | Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW |

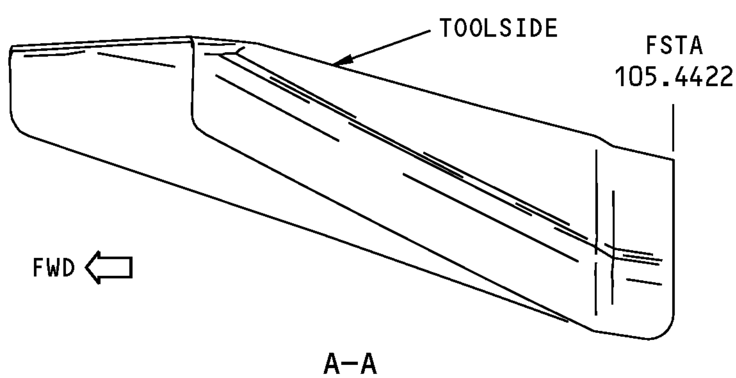
**NOTE**: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.



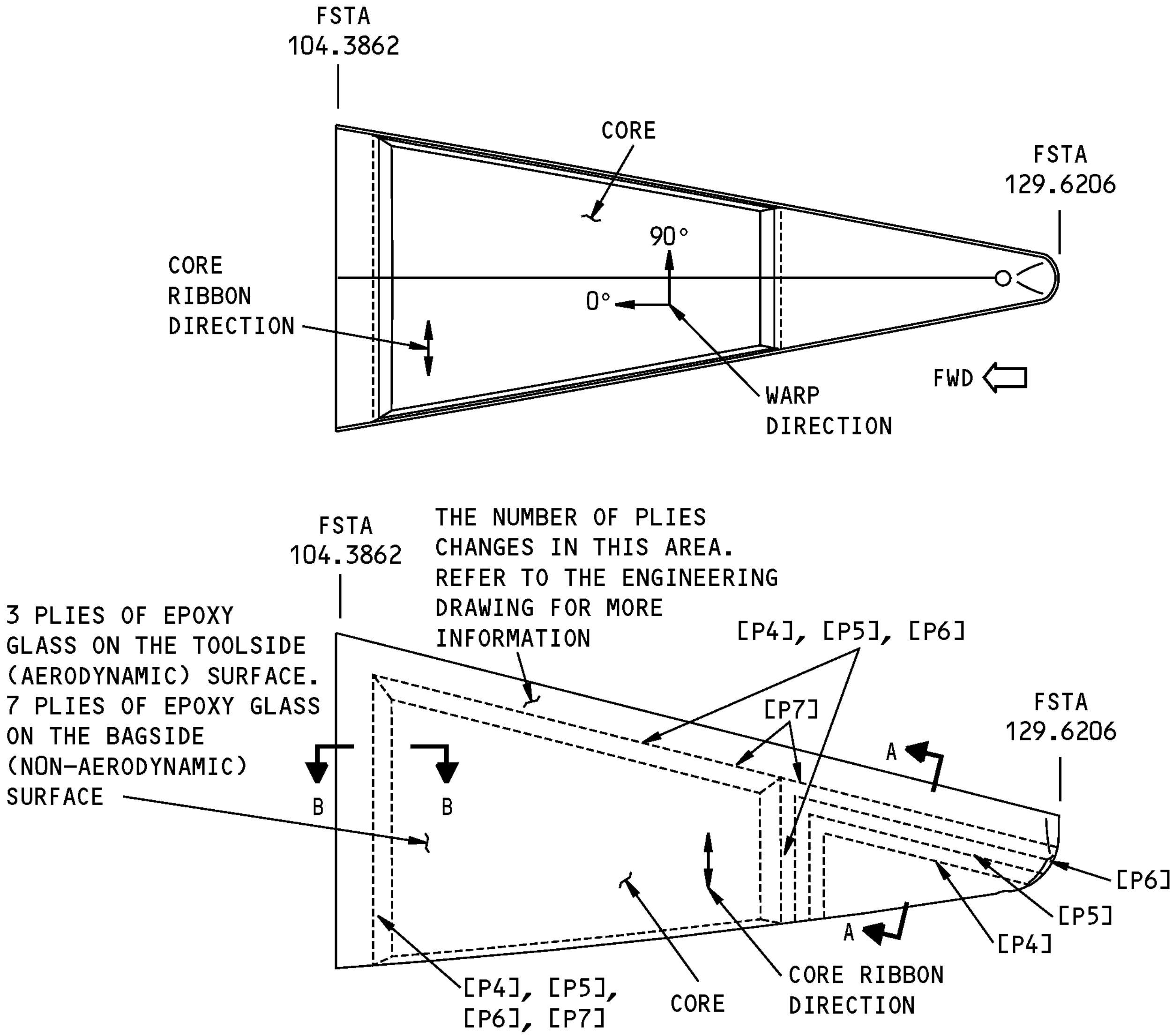




**Warp Direction of Material for Bonded Access Door, Figure 4, Item [2]**
**Figure 8**




**Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [3]**
**Figure 9**



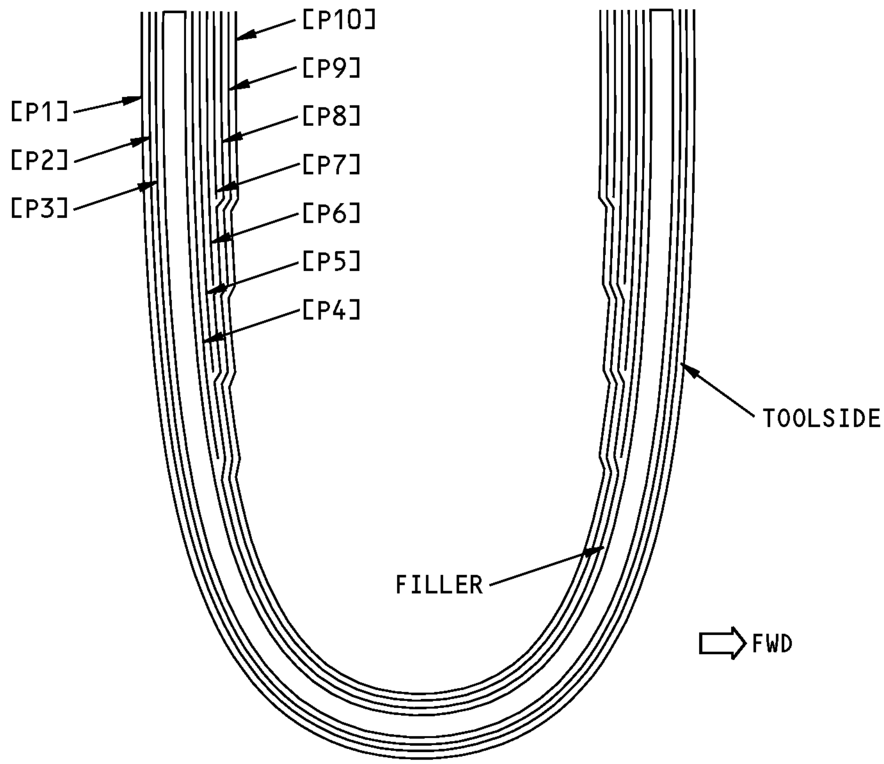

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

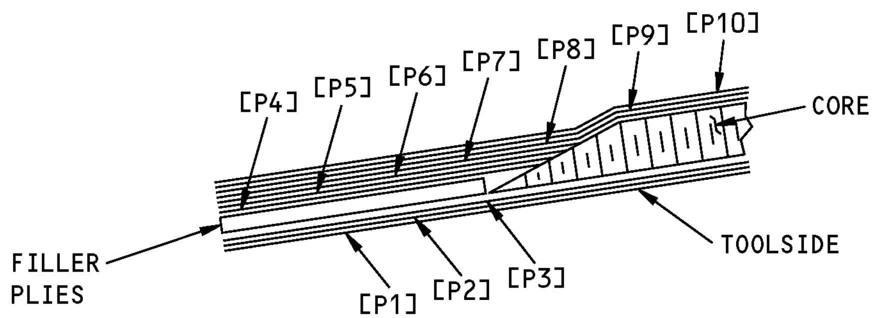
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 10 (Sheet 1 of 2)**




A-A

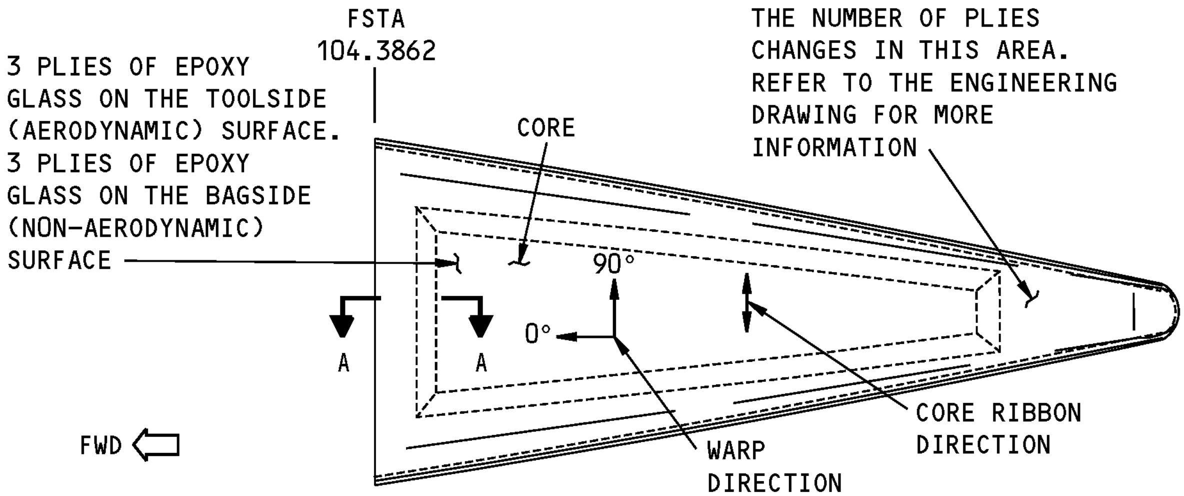



B-B

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 10 (Sheet 2 of 2)**

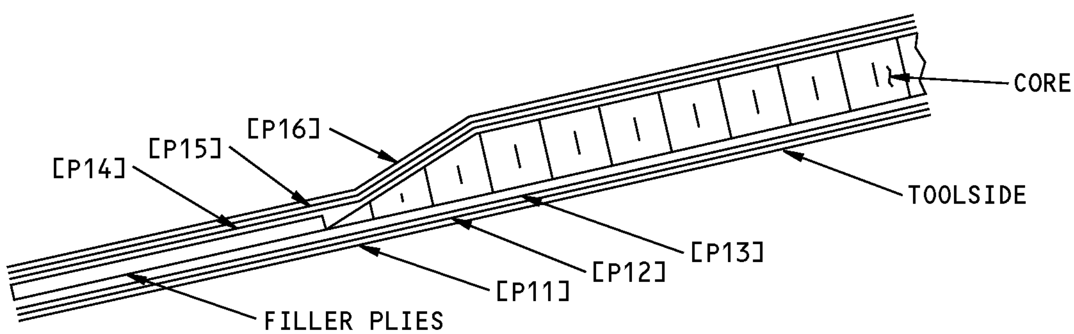
**Table 7**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 10 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 thru P3, and P8 thru P10 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220) |
| P4 thru P7 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781) |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

NOTES

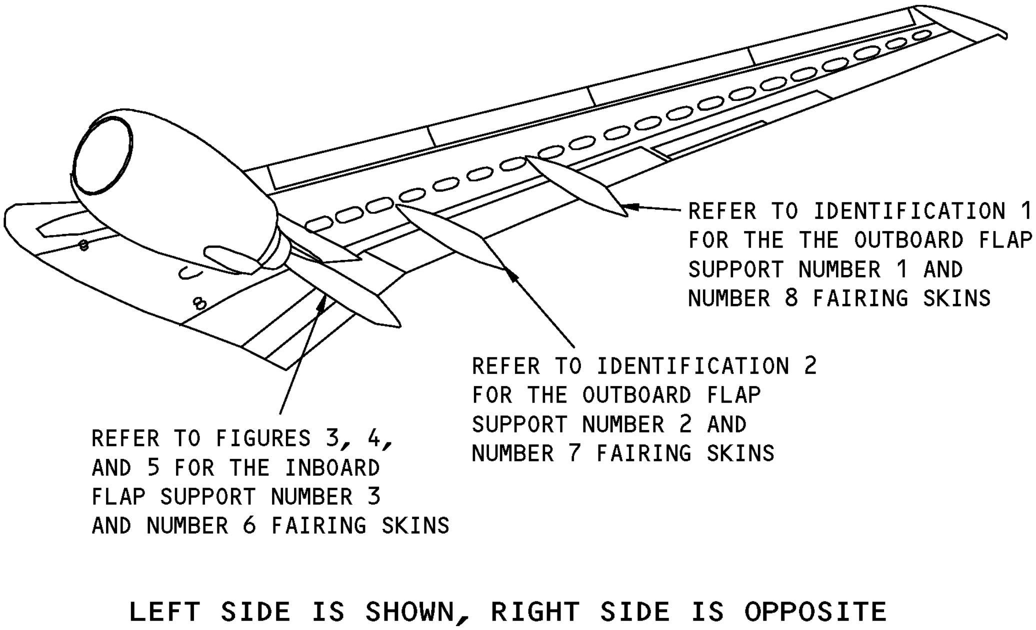
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]**
**Figure 11**

**Table 8**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 11 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P11 thru P16 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781) |

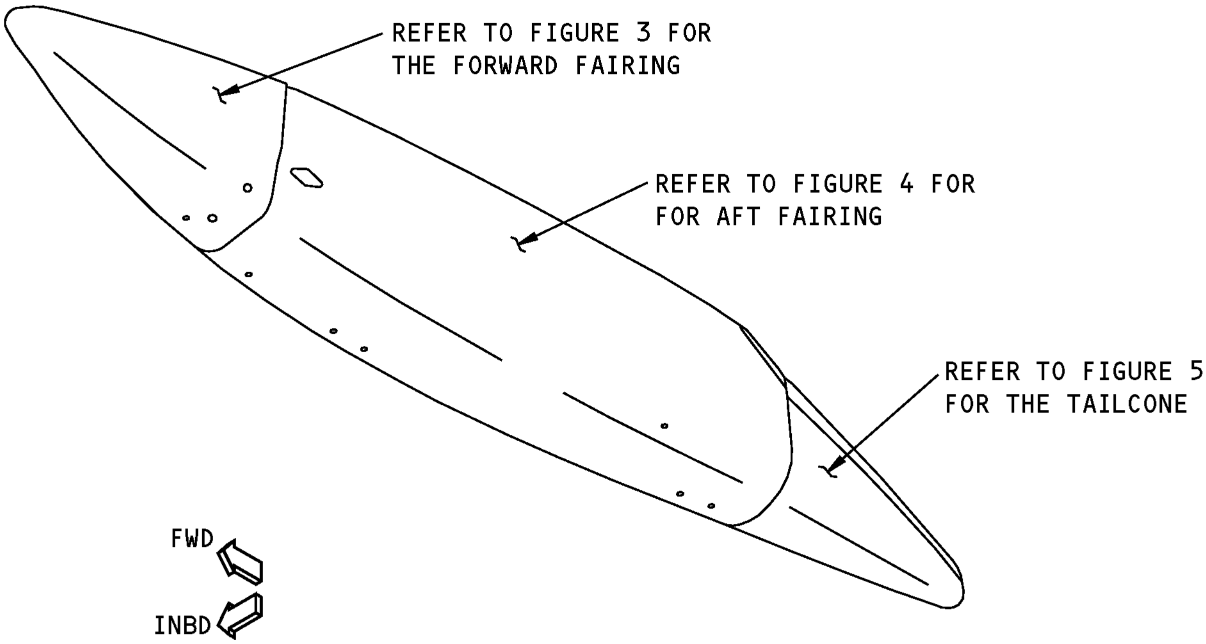
## IDENTIFICATION 3 - FAIRING SKINS - WING INBOARD TRAILING EDGE FLAP SUPPORTS - NUMBERS 3 AND 6




**Wing Trailing Edge Flap Support Fairing Locations**
**Figure 1**

**Table 1**:

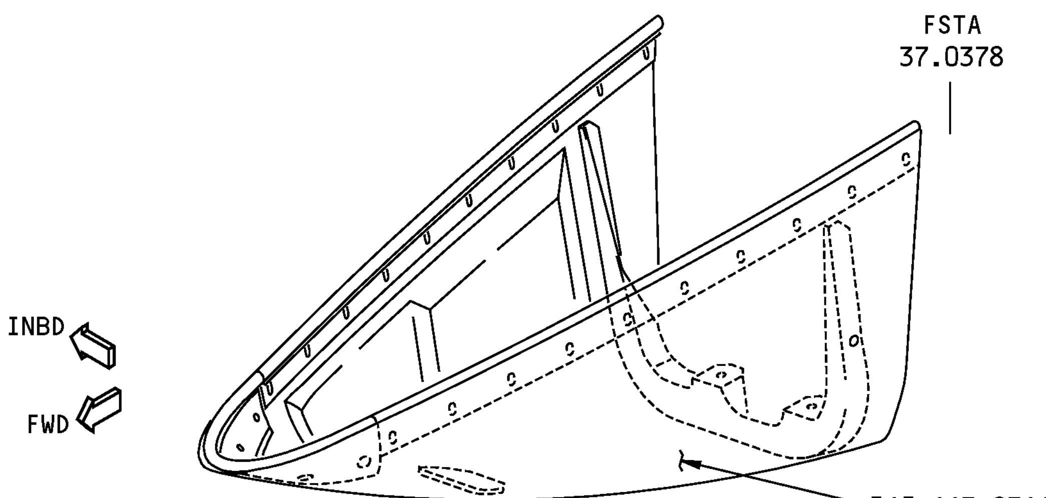
| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A0001 | Movable Trailing Edge Functional Product Collector |
| 113A9300 | Fairing Installation - Flap Support Numbers 3 and 6, Inboard Trailing Edge Flap |
| 113A9310 | Forward Fairing Assembly - Flap Support Number 3, Inboard Trailing Edge Flap |
| 113A9350 | Aft Fairing Assembly - Flap Support Number 3, Inboard Trailing Edge Flap |
| 113A9352 | Tailcone Assembly - Flap Support Fairing Numbers 3 and 6 |




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

NUMBER 3 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

**Inboard Trailing Edge Flap Support Fairing Skin Numbers**
**Figure 2**




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

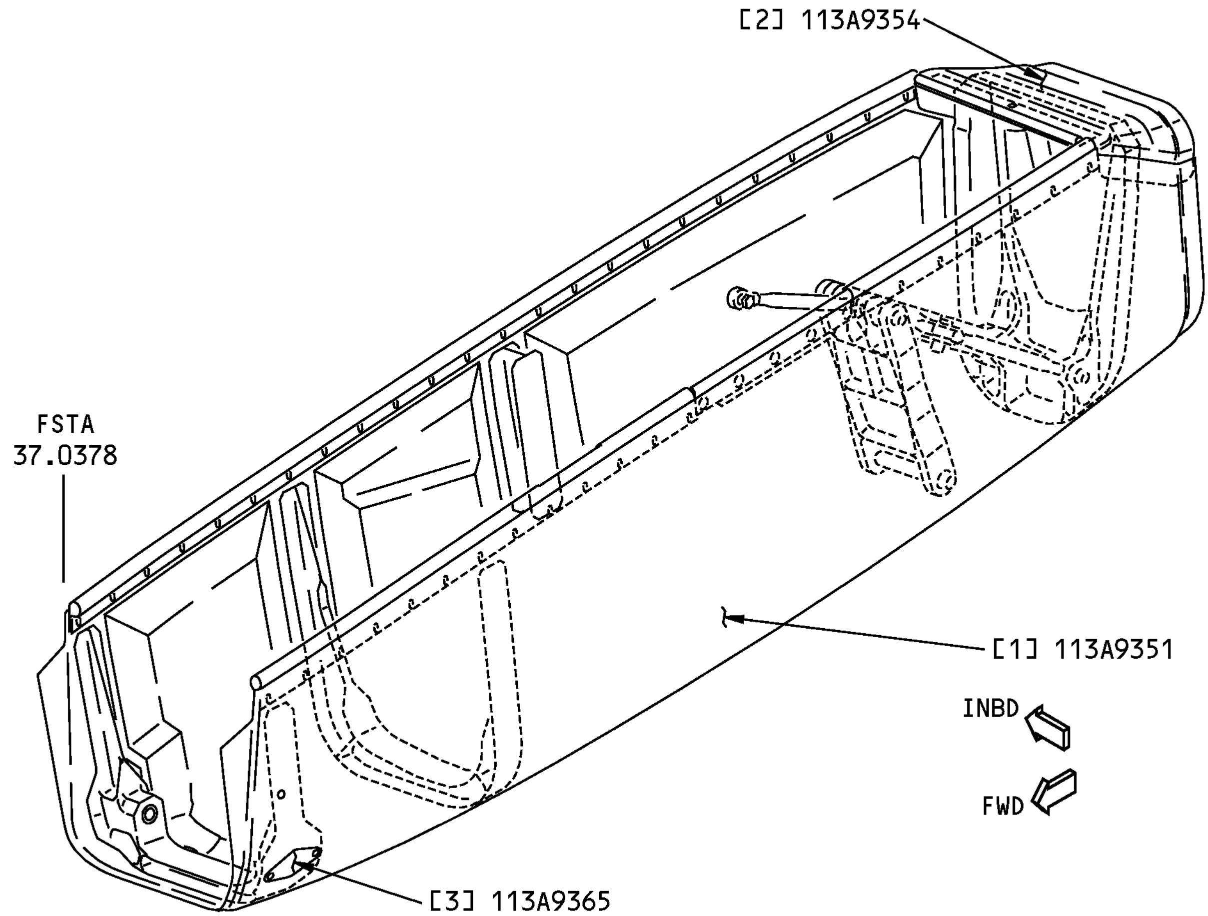
NUMBER 3 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

**Inboard Trailing Edge Flap Support Forward Fairing Skin Identification**
**Figure 3**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Bonded Skin Assembly | | Glass Fiber Reinforced Plastic (GFRP) honeycomb sandwich | |
| | Skin | | Refer to Figure 6 | |
| | Core (2) | 0.50 (12.7)<br>0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

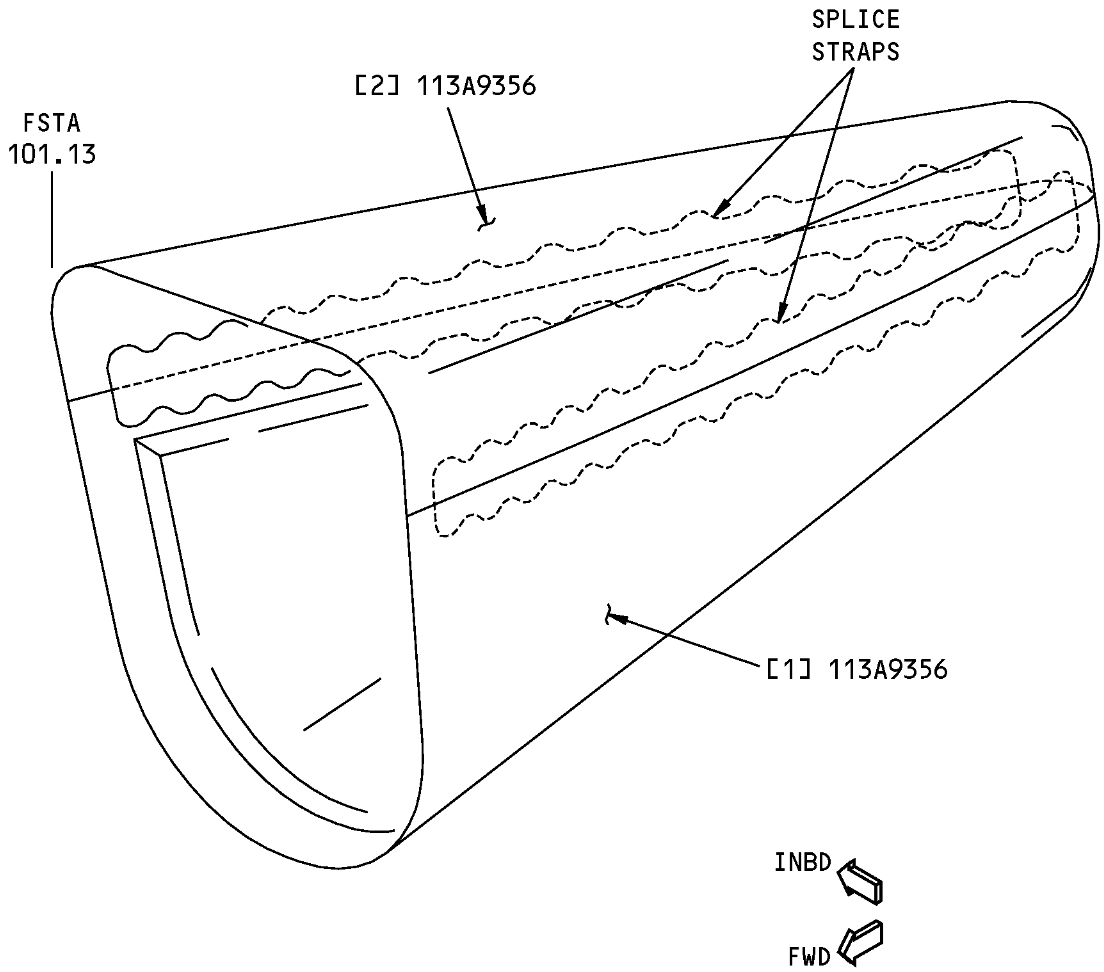
NUMBER 3 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

**Inboard Trailing Edge Flap Support Aft Fairing Skin Identification**
**Figure 4**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Bonded Skin Assembly<br>Skin<br>Core (2) | <br><br>0.75 (19.0) | GFRP honeycomb sandwich<br>Refer to Figure 7<br>Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [2] | Bonded Aft Fairing Cap | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120 (Optional: Style 7781) | Airplane line numbers 1 thru 2187 and does not have SB-737–57–1288 Incorporated<br>Refer to Figure 8 |
| [2] | Bonded Aft Fairing Cap | | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120 | Airplane line numbers 2188 and on, or airplanes that have SB-737–57–1288 incorporated<br>Refer to Figure 9 |
| [3] | Bonded Access Cover (2) | | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

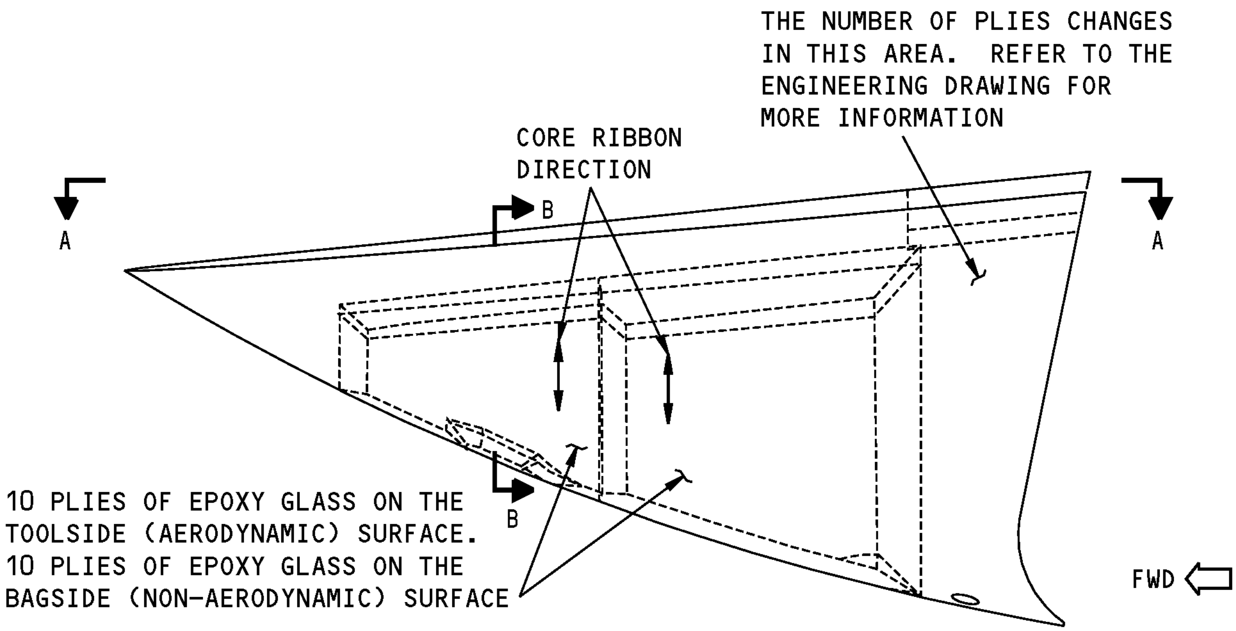
NUMBER 3 FLAP SUPPORT TAILCONE FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT TAILCONE FAIRING (RIGHT SIDE) IS OPPOSITE

**Inboard Trailing Edge Flap Support Tailcone Fairing Skin Identification**
**Figure 5**

**Table 4**:

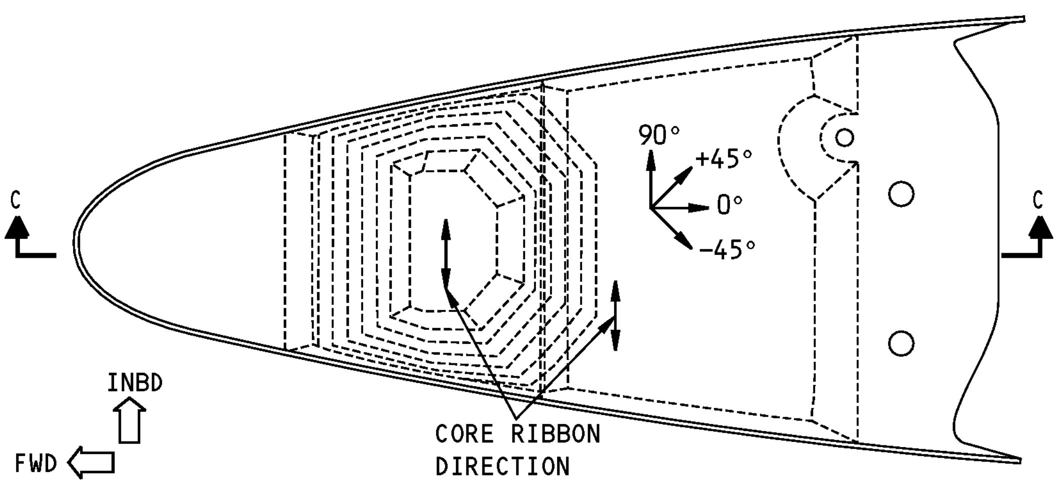
| LIST OF MATERIALS FOR FIGURE 5 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Lower Bonded Tailcone Assembly | | GFRP honeycomb sandwich | |
| | Skin | | Refer to Figure 1 | Airplane line numbers 1 thru 2187 and does not have SB-737–57–1288 Incorporated |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [1] | Lower Bonded Tailcone Assembly | | GFRP honeycomb sandwich | |
| | Skin | | Refer to Figure 11 | Airplane line numbers 2188 and on, or airplanes that have SB-737–57–1288 incorporated |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [2] | Upper Bonded Tailcone Assembly | | GFRP honeycomb sandwich | |
| | Skin | | Refer to Figure 12 | Airplane line numbers 1 thru 2187 and does not have SB-737–57–1288 Incorporated |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |
| [2] | Upper Bonded Tailcone Assembly | | GFRP honeycomb sandwich | |
| | Skin | | Refer to Figure 13 | Airplane line numbers 2188 and on, or airplanes that have SB-737–57–1288 incorporated |
| | Core | 0.25 (6.4) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

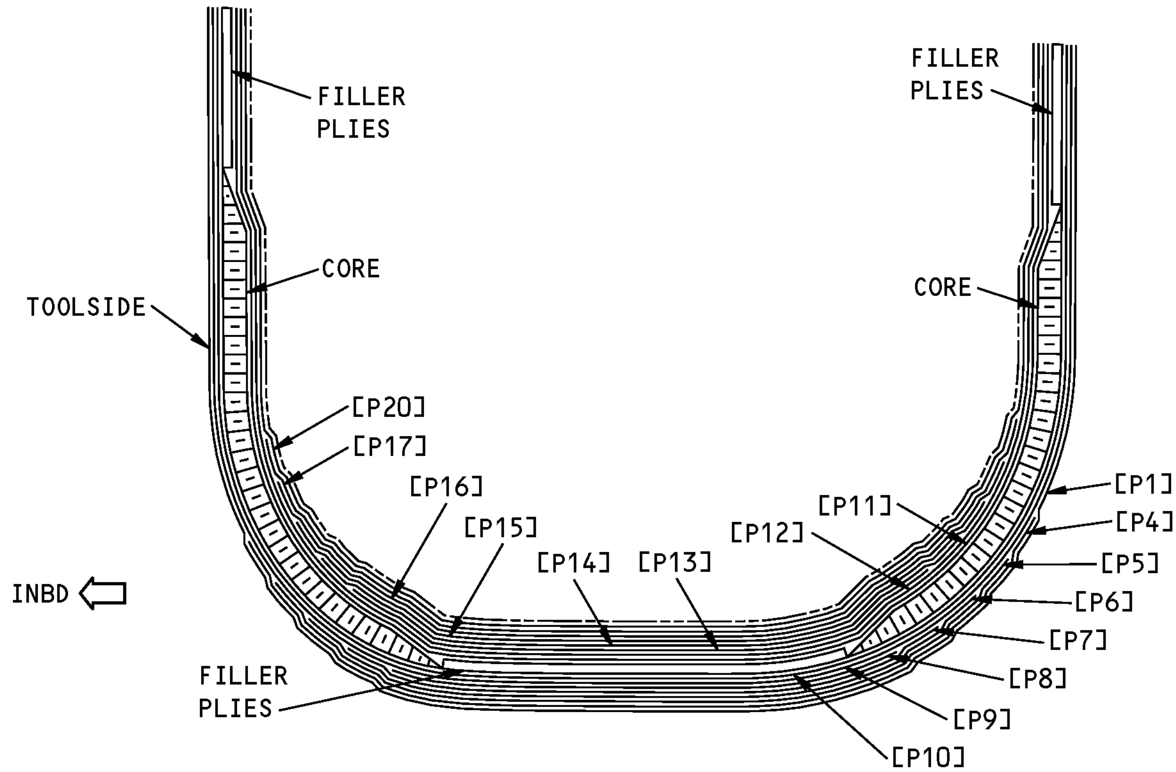



A-A

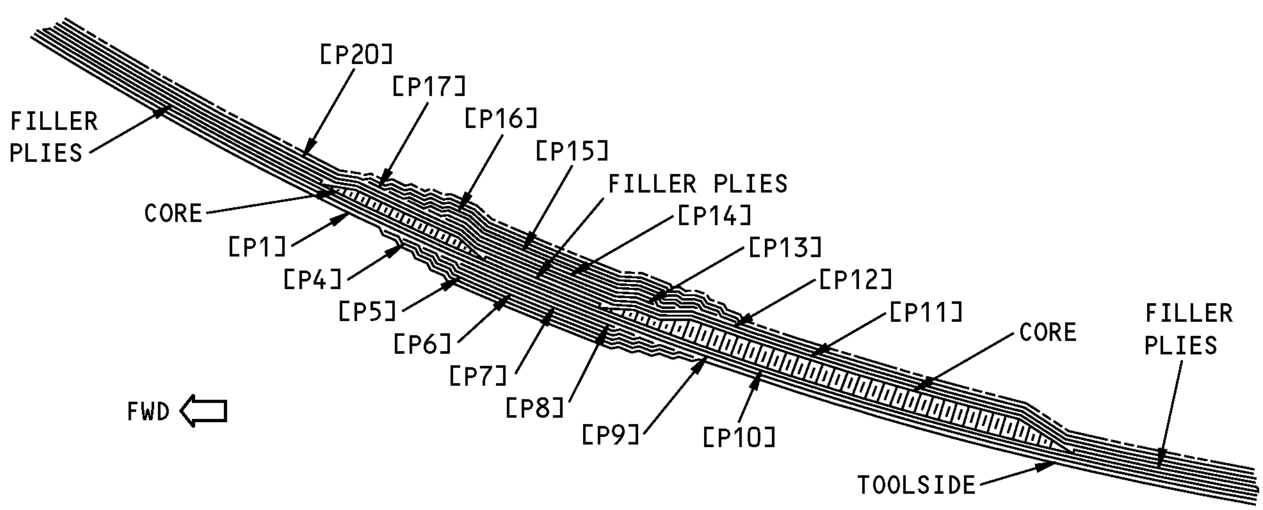
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO SECTION A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B AND C-C FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]**
**Figure 6 (Sheet 1 of 2)**




B-B



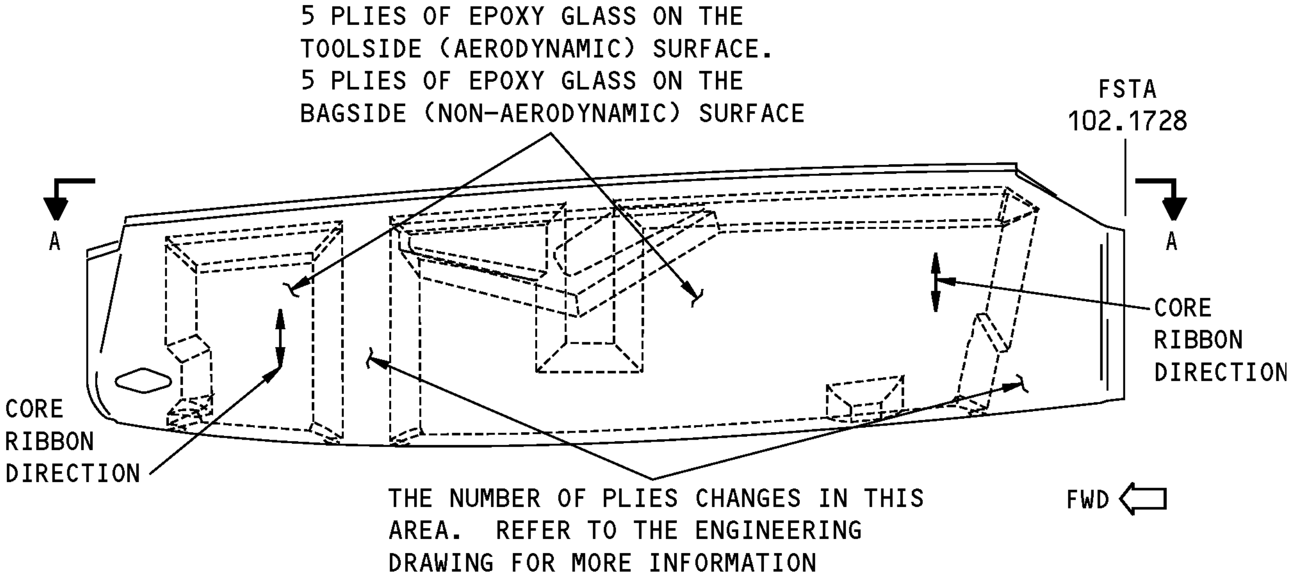

C-C

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]**
**Figure 6 (Sheet 2 of 2)**

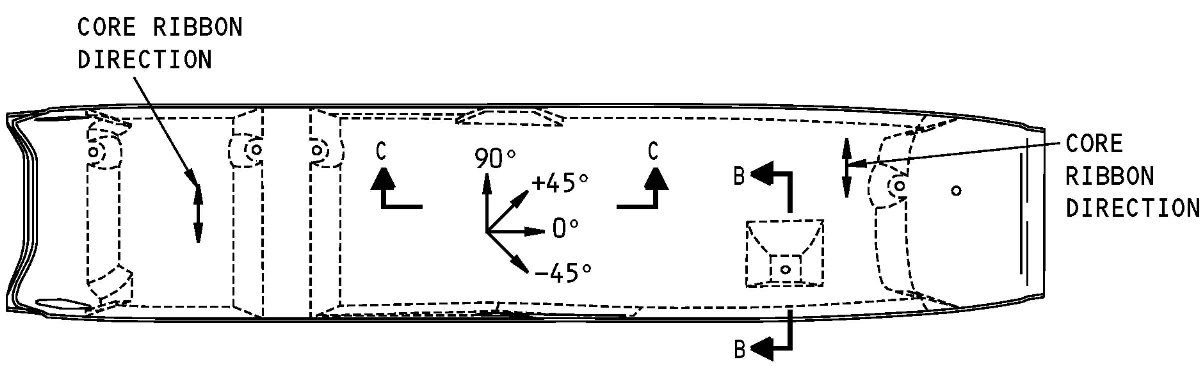
**Table 5**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 6 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P10, P11, and P20 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120. (Optional: Style 220) |
| P2 thru P4, P6, P8, P13, P15, and P17 thru P19 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| P5, P7, P9, P12, P14, and P16 | + or - 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220) |

**NOTE**: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION
A

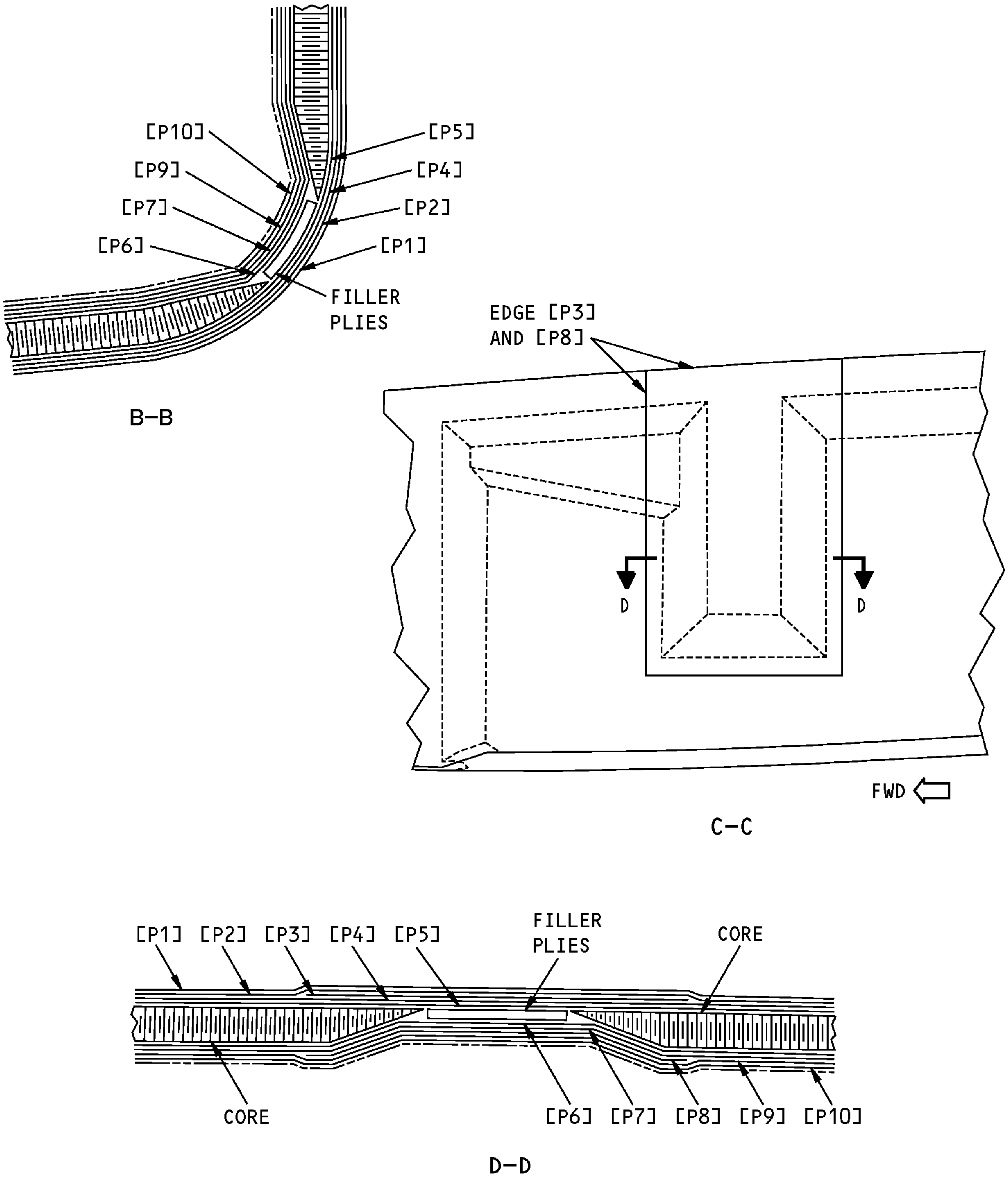



A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO SECTION A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B, C-C, AND D-D FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 1 of 2)**

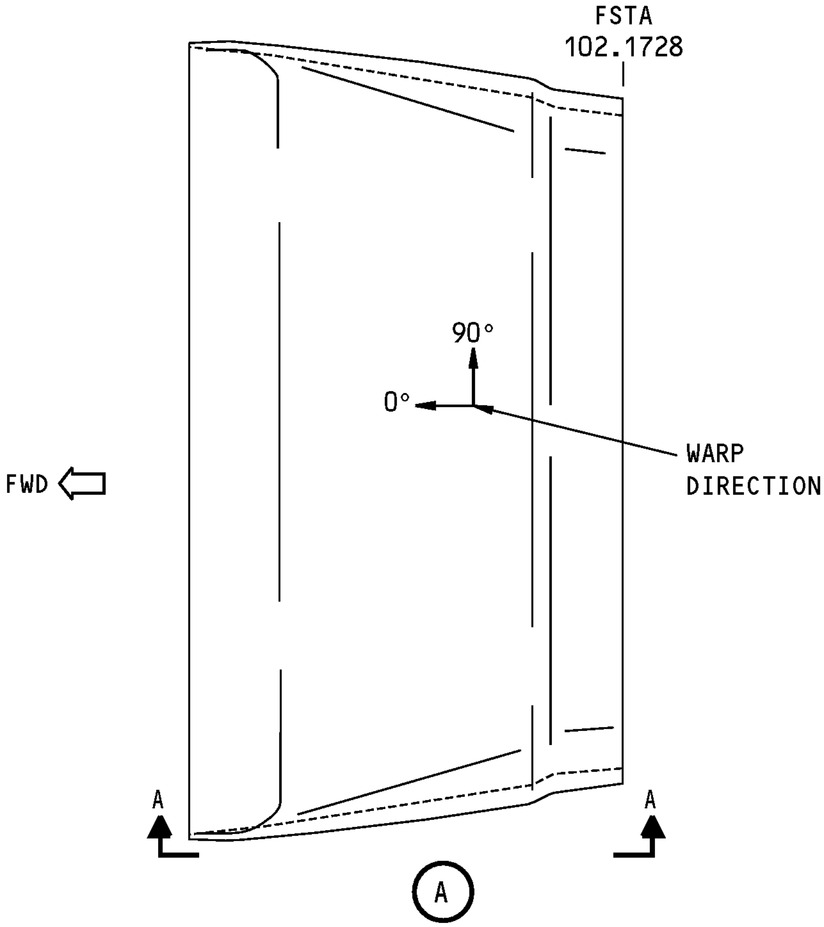


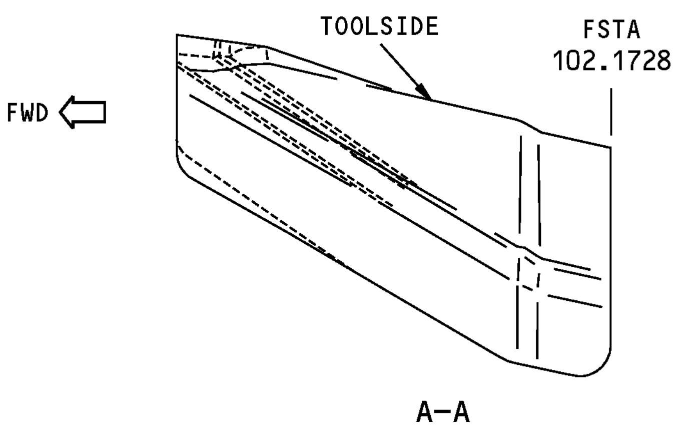

**Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]**
**Figure 7 (Sheet 2 of 2)**

Table 6:

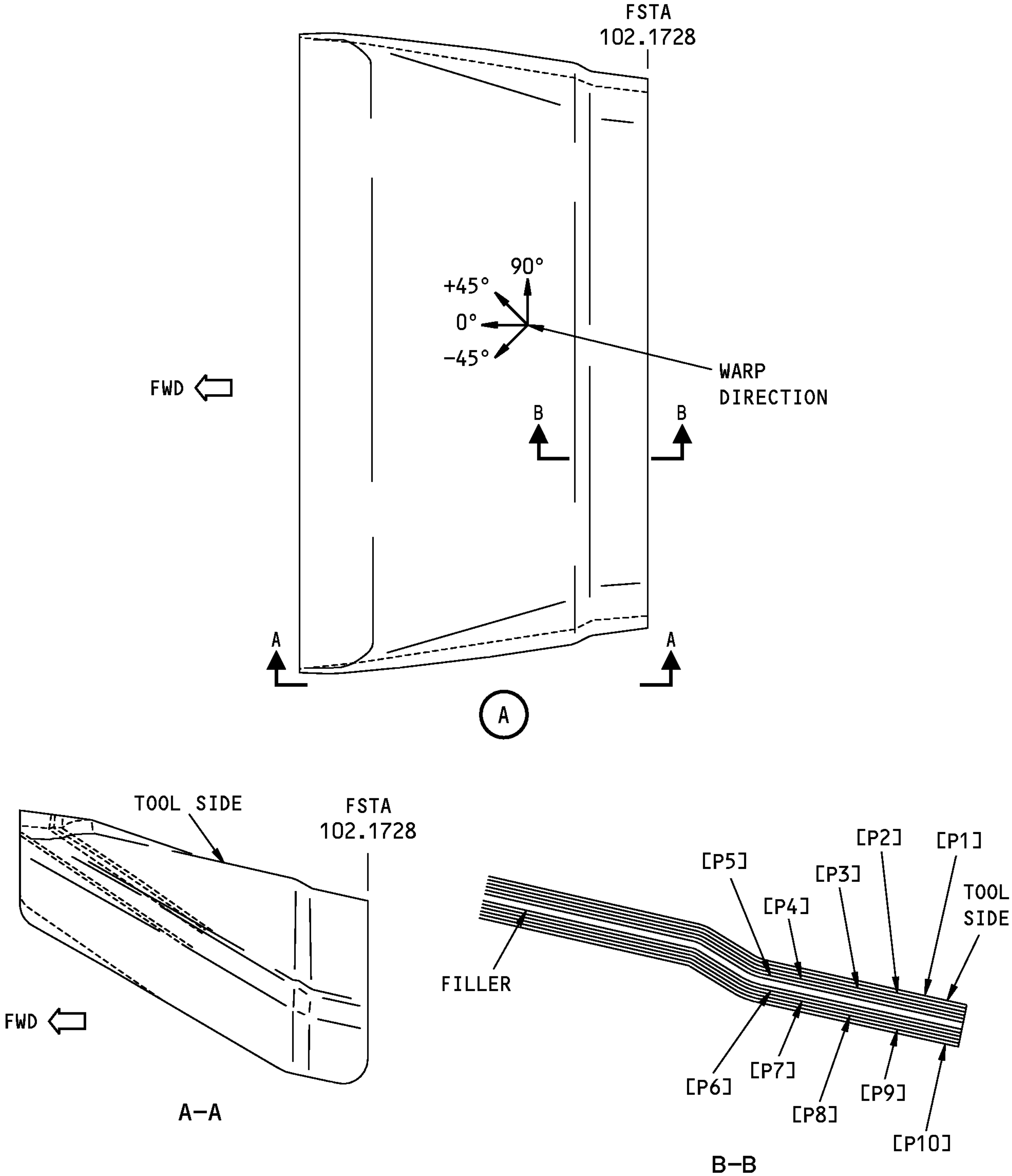
| PLY MATERIAL AND DIRECTION FOR FIGURE 7 ITEM [1] | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1, P5, P6, and P10 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 120. (Optional: Style 220) |
| P2, P4, P7, and P9 | + or - 45 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 (Optional: Style 7781) |
| P3, and P8 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581. (Optional: Style 7781) |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120 (Optional: Style 7781) |

**NOTE**: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.



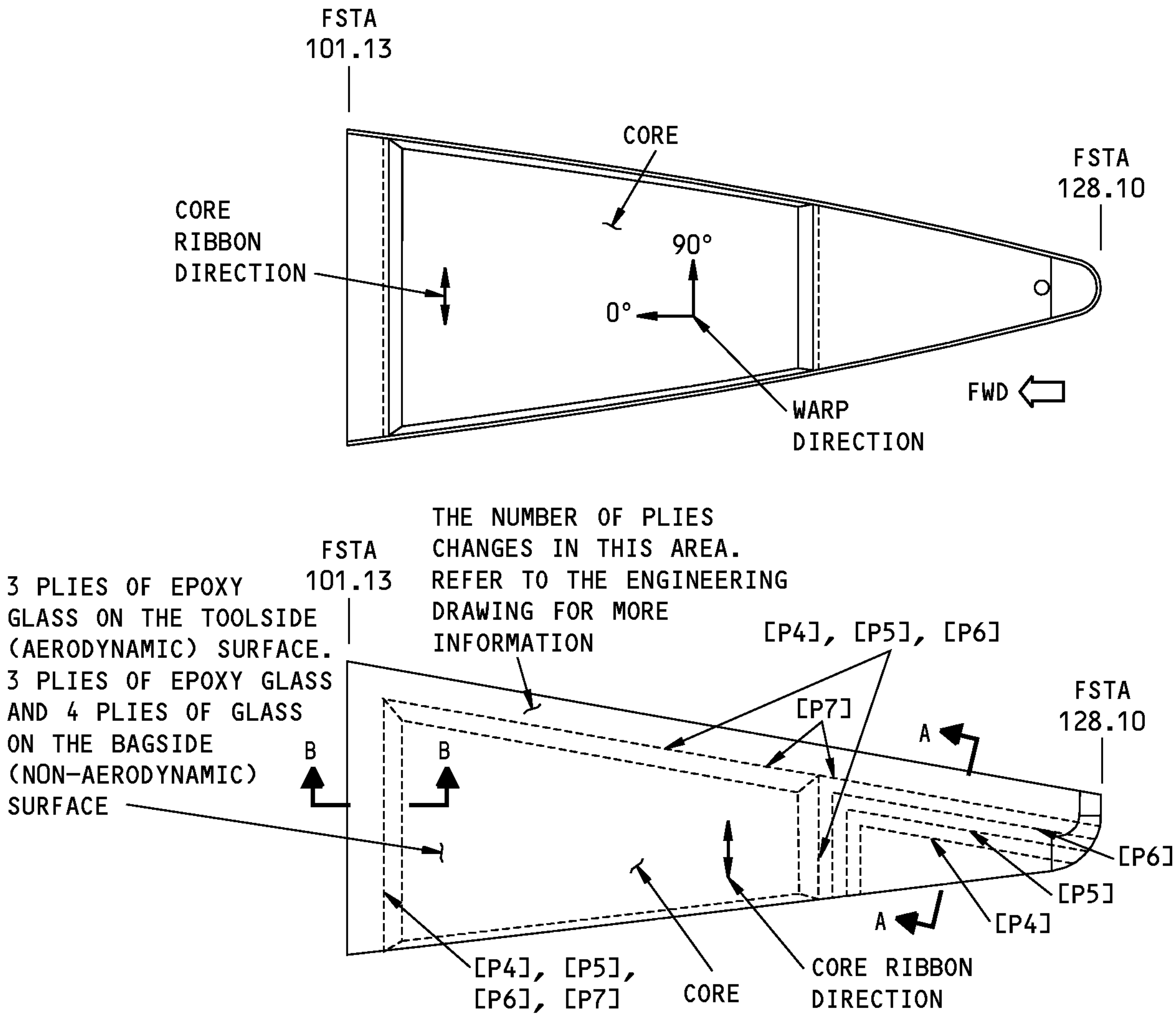

**Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [2]**
**Figure 8**




**Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [2]**
**Figure 9**

**Table 7**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 9 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P5, P6, P8, and P10 | 0 or 90 degrees | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |
| P2, P4, P7, and P9 | + or - 45 degrees | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |
| Filler Plies | Optional | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |



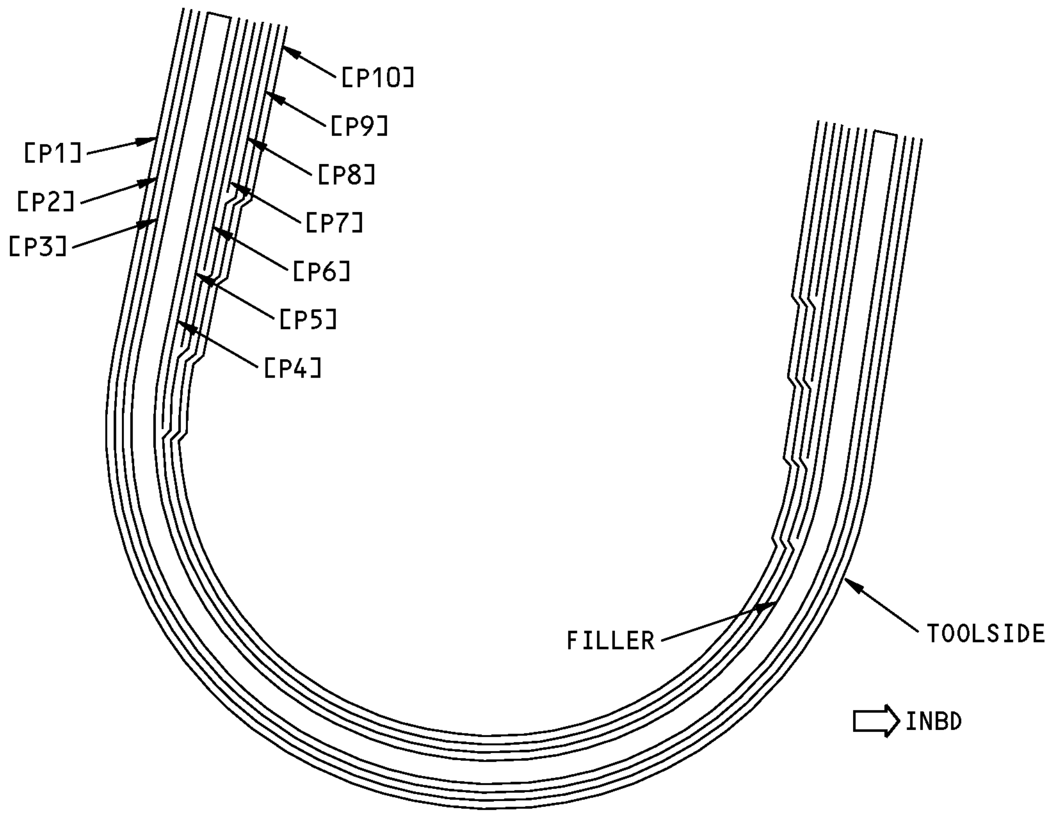

VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

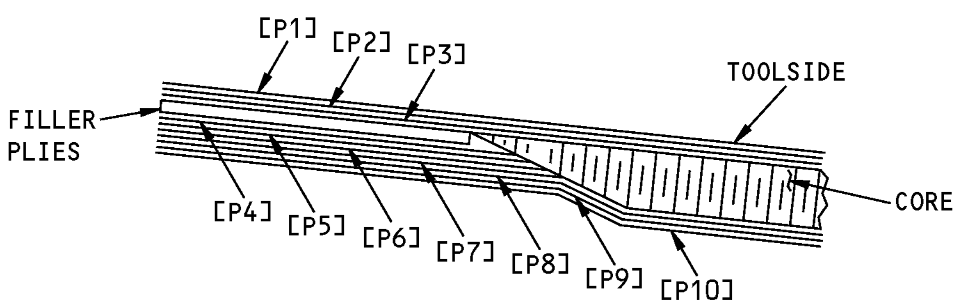
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 10 (Sheet 1 of 2)**




A-A

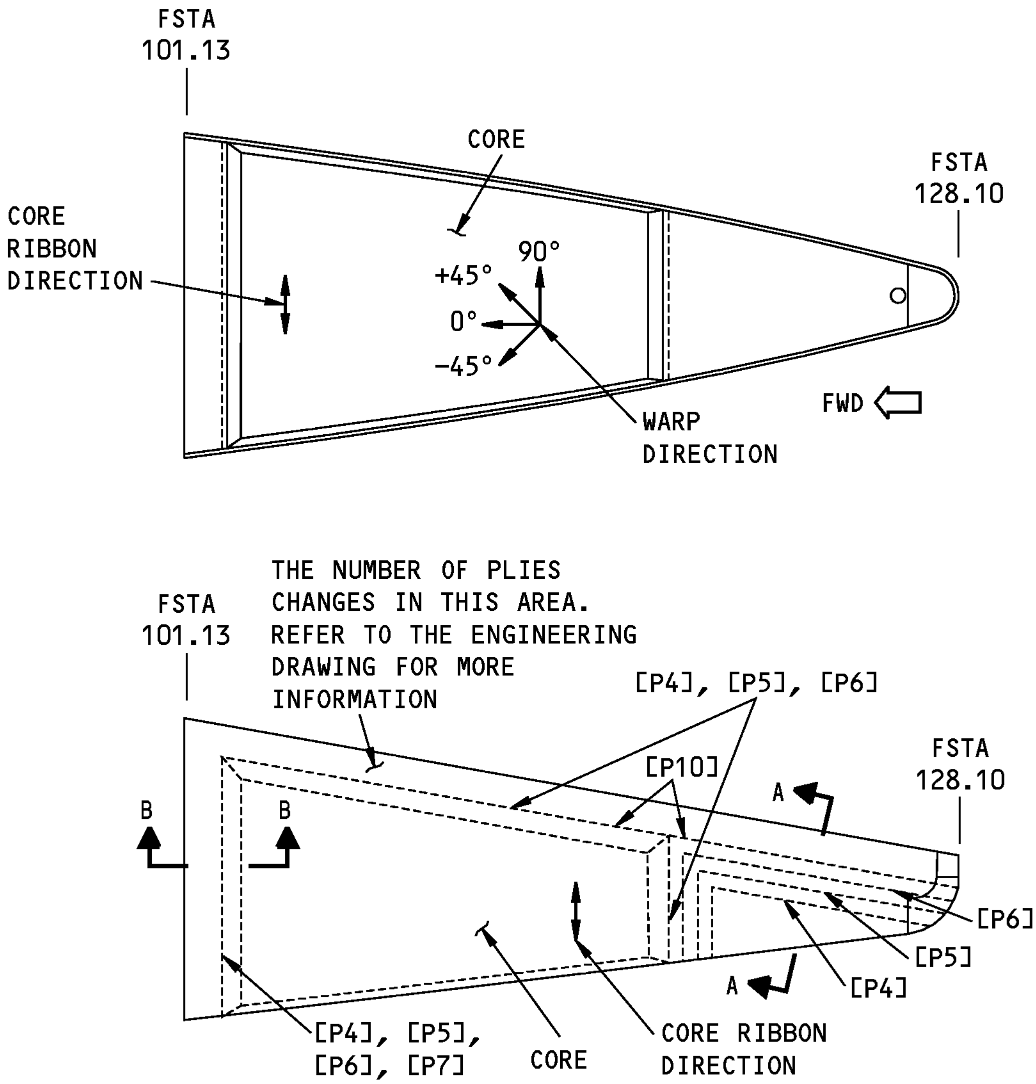



B-B

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 10 (Sheet 2 of 2)**

**Table 8**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 10 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 thru P3, and P8 thru P10 | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 120 |
| P4 thru P7 | 0 or 90 degrees | Glass fabric as given in BMS 8-139, Class I, Style 120 |
| Filler Plies | 0 or 90 degrees | Glass fabric as given in BMS 8-139, Class I, Style 120 |



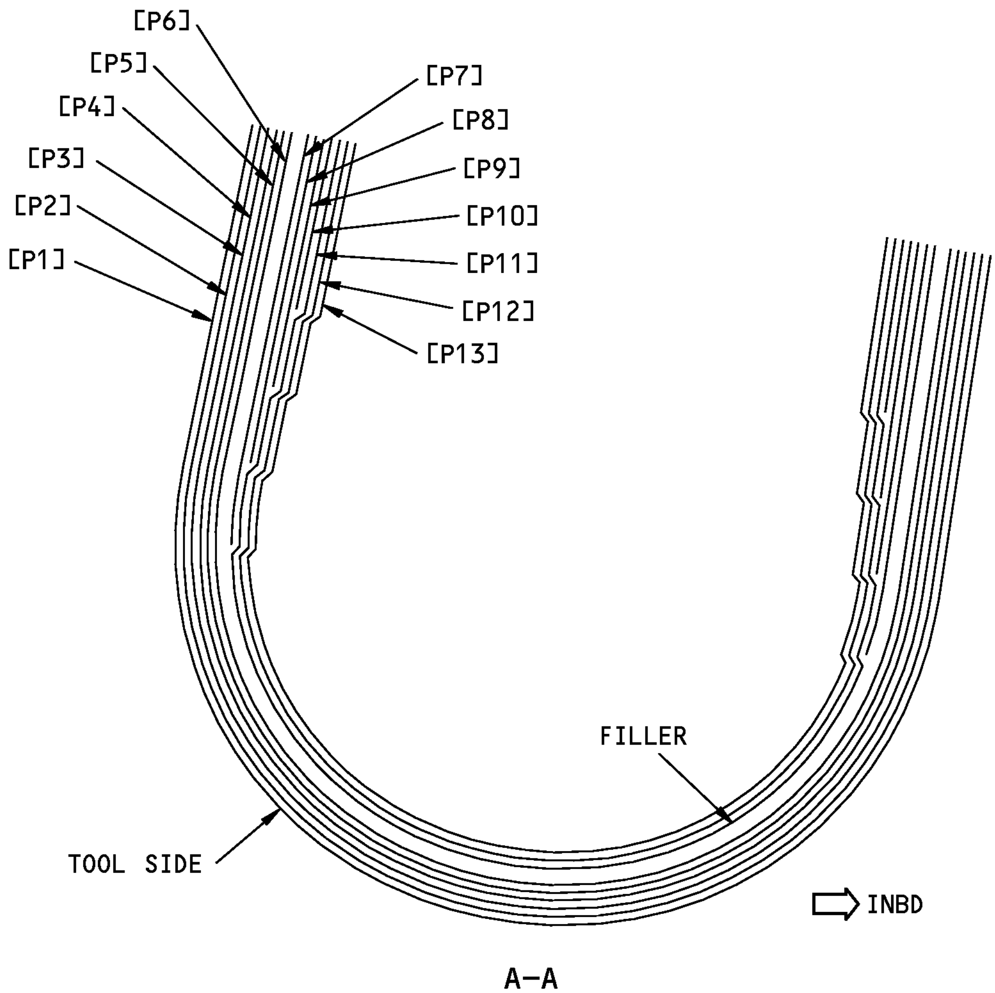

VIEW IS ON THE TOOL SIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

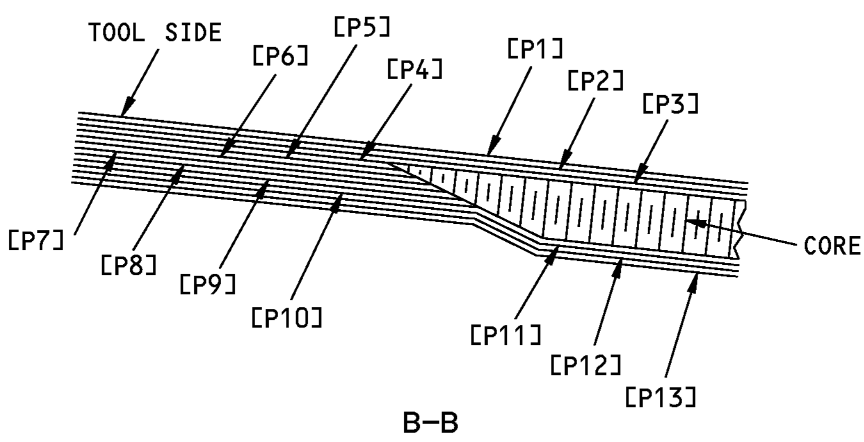
A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 11 (Sheet 1 of 2)**

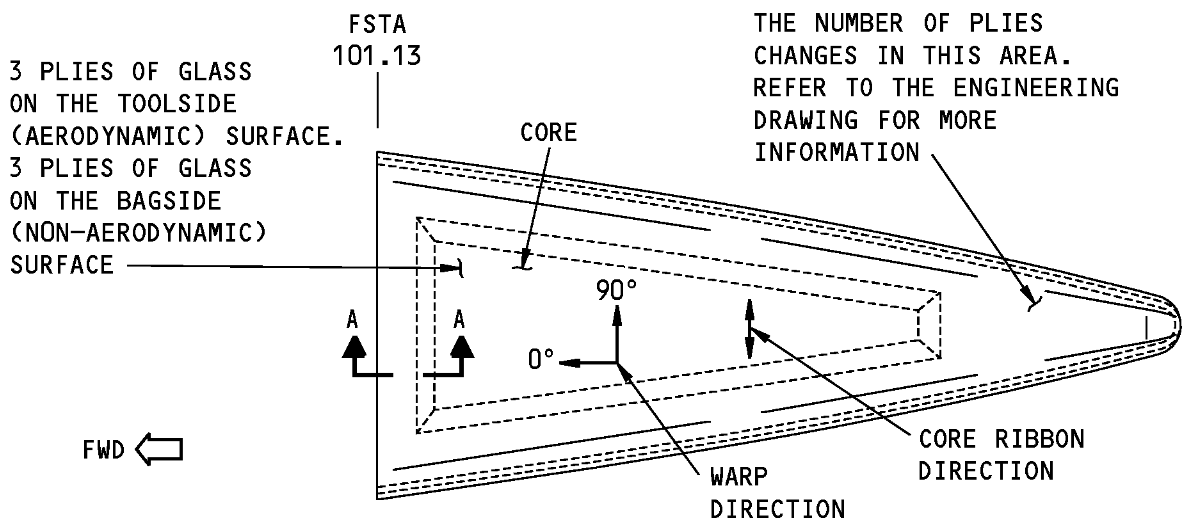



**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]**
**Figure 11 (Sheet 2 of 2)**

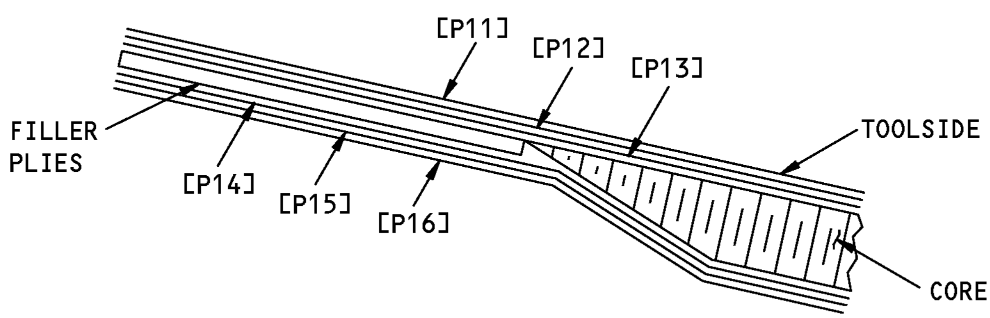
**Table 9**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 11 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P4, P6, P8, P10, and P11 | 0 or 90 degrees | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |
| P13 | + or - 90 degress | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |
| P2, P5, P7, P9, and P12 | + or - 45 degrees | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |
| Filler Plies | Optional | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |




VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

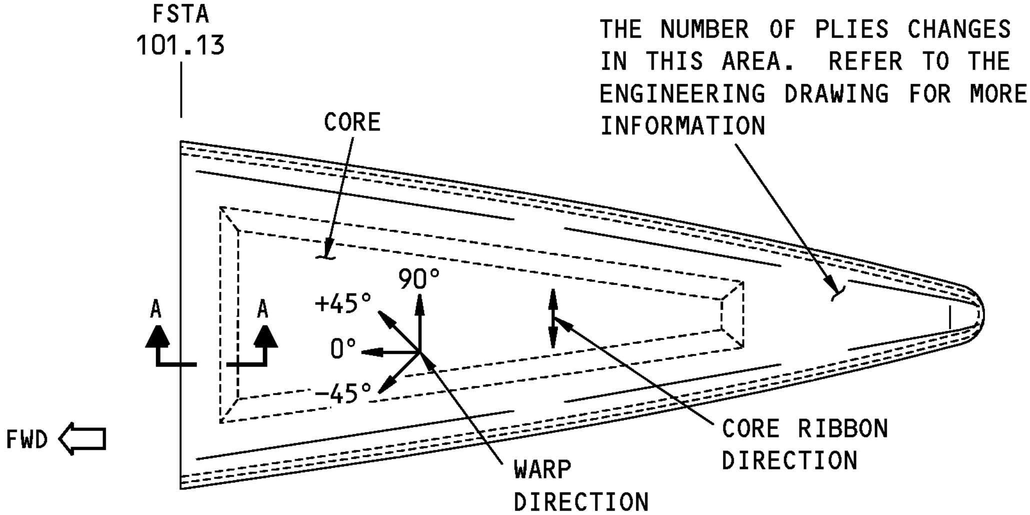
NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]**
**Figure 12**

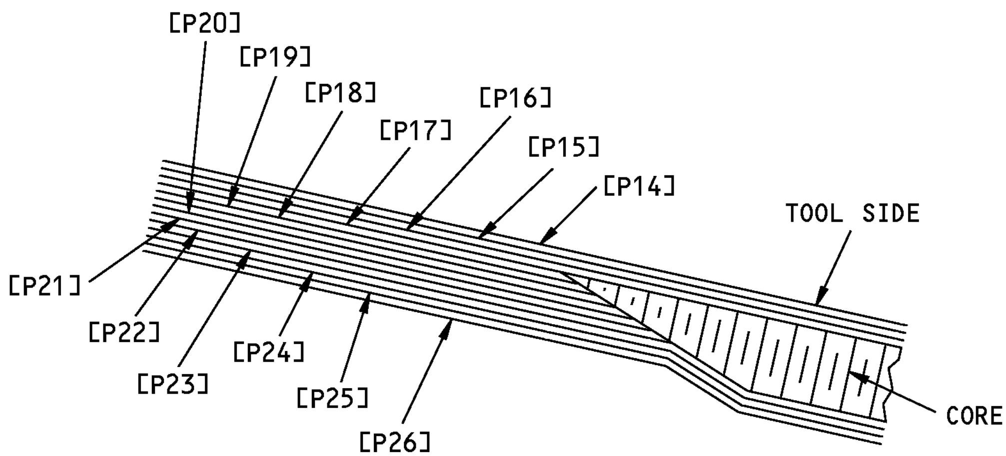
**Table 10**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 12 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P11 thru P16 | 0 or 90 degrees | Glass fabric as given in BMS 8-139, Class I, Style 120 |
| Filler Plies | 0 or 90 degrees | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120 (Optional: Style 7781) |




VIEW IS ON THE TOOL SIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 11 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]**
**Figure 13**

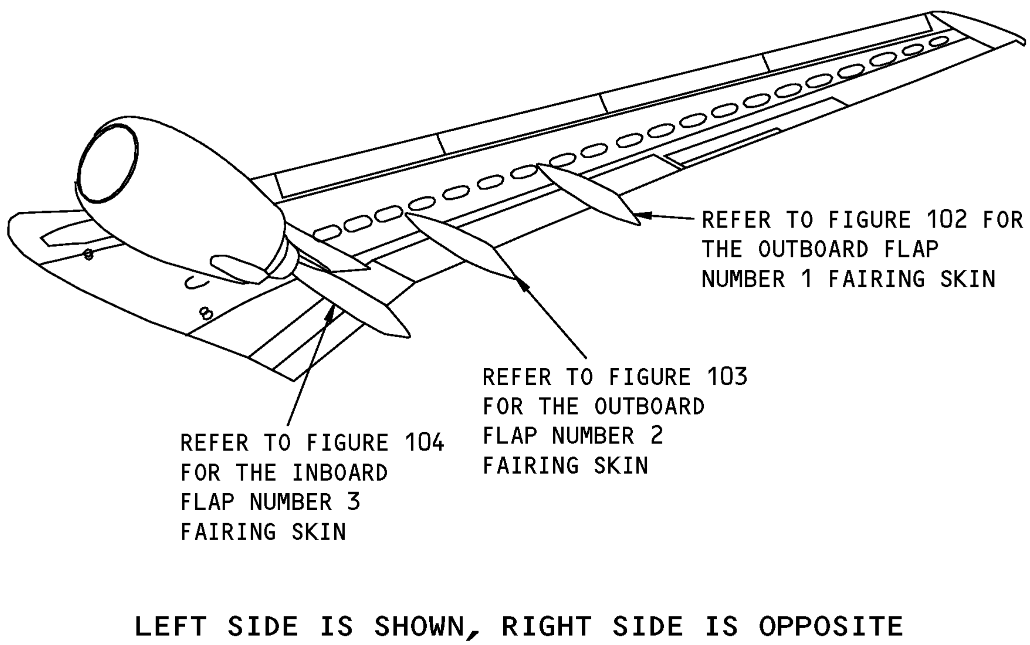
**Table 11**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 13 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P14, P16, P17, P19, P21, P23, P24, and P26 | 0 or 90 degrees | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |
| P15, P18, P20, P22, and P25 | + or - 45 degrees | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |
| Filler Plies | Optional | Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581 |

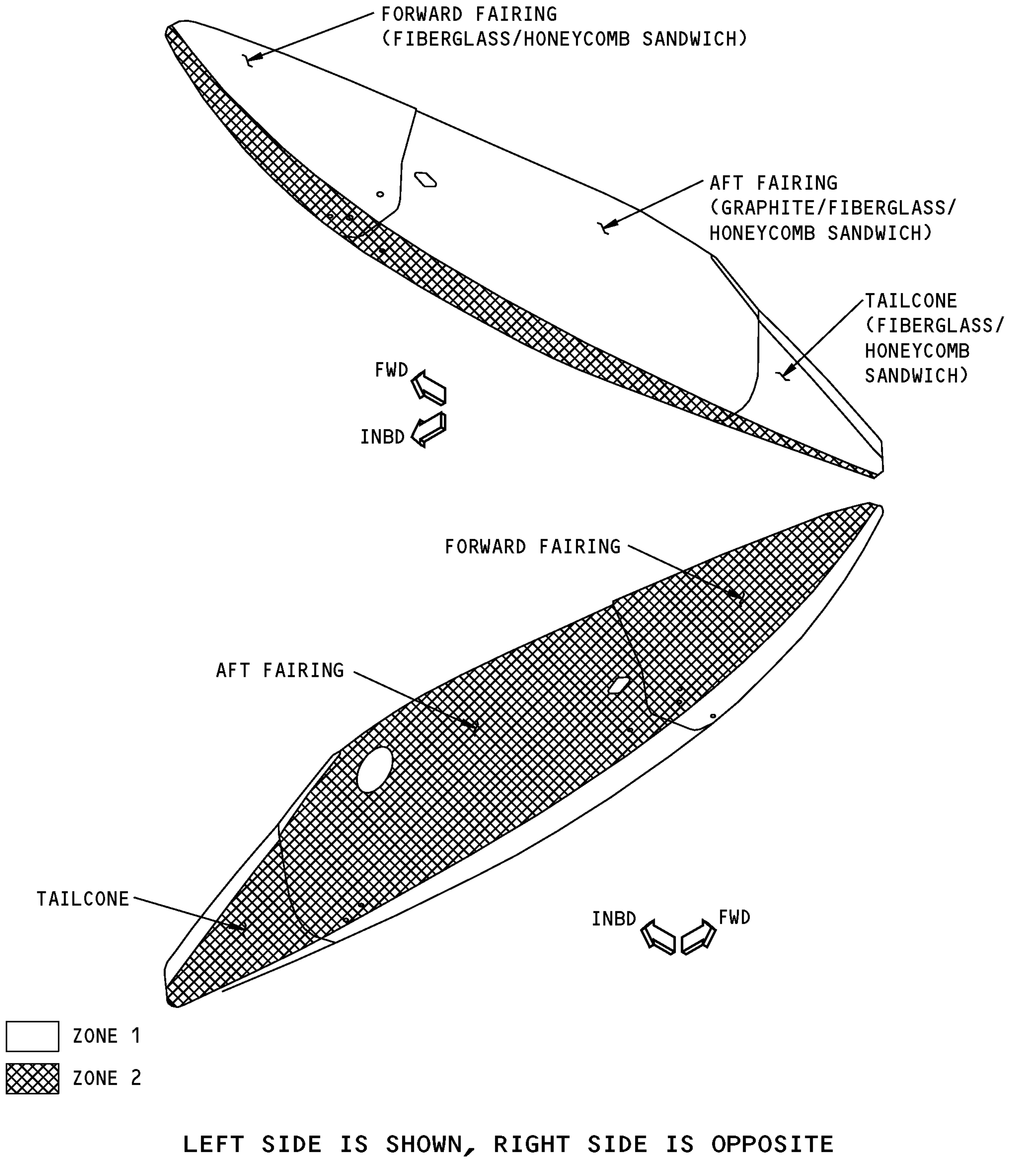
## ALLOWABLE DAMAGE 1 - FLAP SUPPORT FAIRING SKIN, WING TRAILING EDGE FLAP

1. **Applicability**

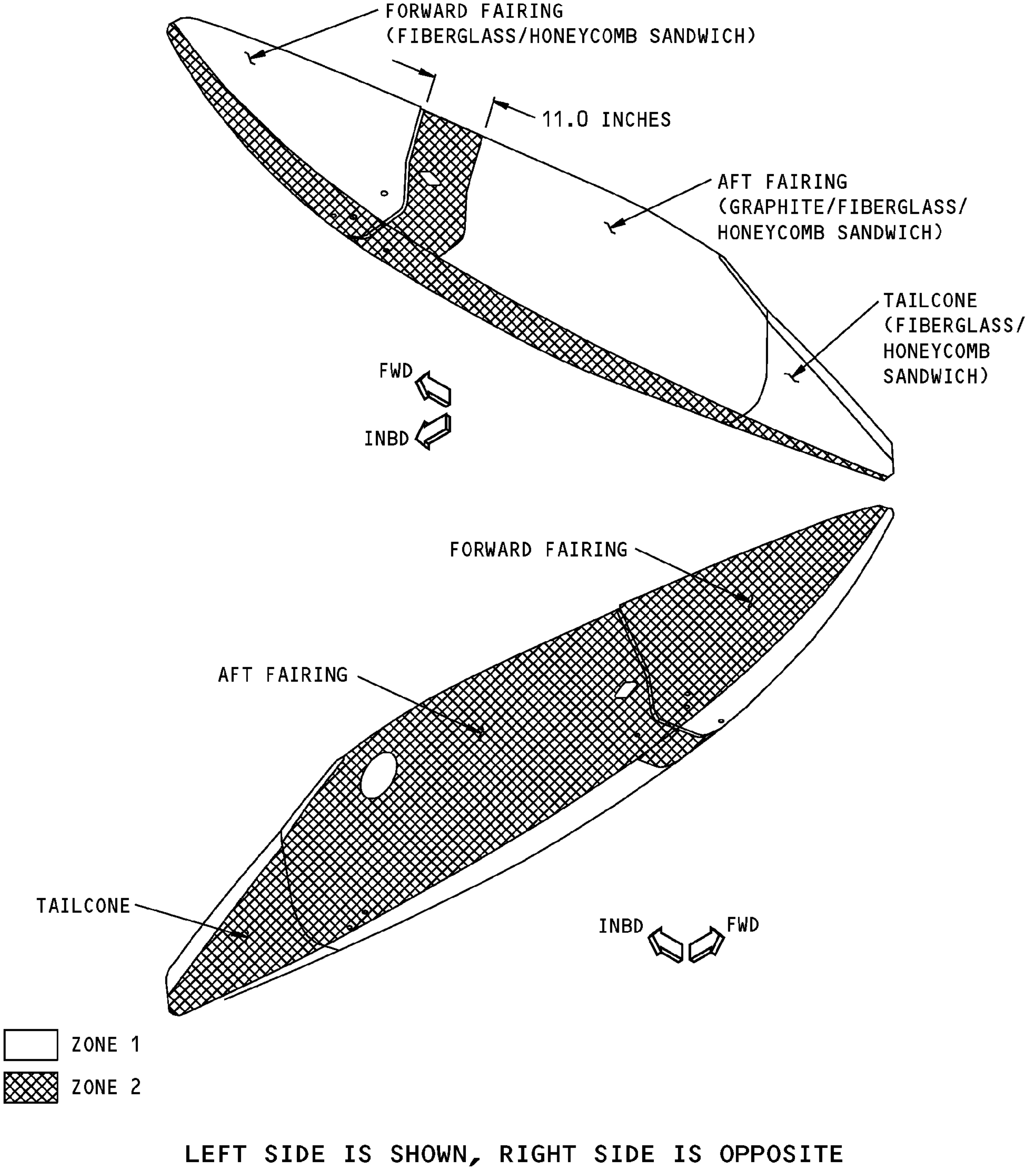
   A. This subject gives the allowable damage limits for the flap support fairing skin of the wing trailing edge flap shown in Flap Support Fairing Location for the Wing Trailing Edge Flap, Figure 101/ALLOWABLE DAMAGE 1.



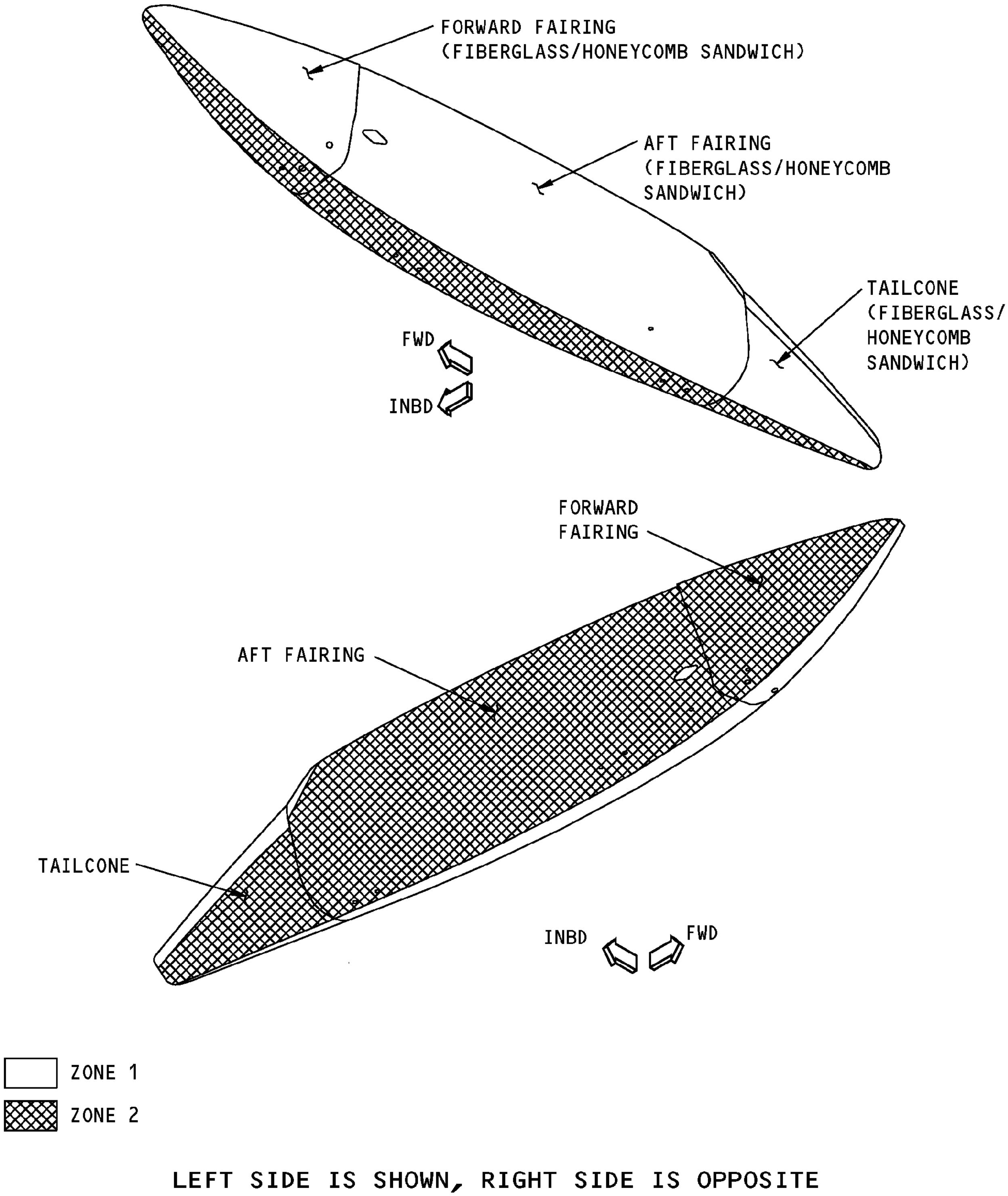

**Flap Support Fairing Location for the Wing Trailing Edge Flap**
**Figure 101**




**Flap Support Fairing Number 1 For the Outboard Trailing Edge Flap**
**Figure 102**




**Flap Support Fairing Number 2 For the Outboard Trailing Edge Flap**
**Figure 103**




**Flap Support Fairing Number 3 for the Inboard Trailing Edge Flap**
**Figure 104**

## 2. General

A. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other approved inspection methods can be used.

(1) Refer to Damage Definitions, Figure 105/ALLOWABLE DAMAGE 1 for the definitions of the length, width, depth, and diameter of damage.

(2) The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause fiberglass or graphite damage and are sealed.

B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two methods that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) If the tape is on an external surface, then do as follows:

1) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

(d) Repair the damage with a permanent seal at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Make a temporary seal with aluminum foil tape (speed tape).

(2) Keep a record of the location.

(3) Repair the damage at or before 400 flight hours from the time the seal was made.

## 3. References

| Reference | Title |
|---|---|
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

(Continued)

| Reference | Title |
|---|---|
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

**4. Allowable Damage Limits**

A. Zone 1 - Honeycomb Core Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the fiberglass or graphite fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the fiberglass or graphite fibers are permitted if they are:

(a) A maximum of one ply in depth.

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail A .

(3) Dents:

(a) Dents that do not cause damage to the carbon glass fibers are permitted if they are:

1) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail A .

**NOTE**: Use the limits for holes and punctures if there is carbon glass fiber damage or if the dent depth is more than 0.05 inch.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail A.

(5) Delamination is permitted if it is:

(a) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail A.

B. Zone 1 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the fiberglass or graphite fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the fiberglass or graphite fibers are permitted if they are:

(a) A maximum of one ply in depth.

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

(3) Dents:

(a) Dents that do not cause damage to the fiberglass or graphite fibers are permitted if they are:

1) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

**NOTE**: Use the limits for holes and punctures if there is carbon glass fiber damage or if the dent depth is more than 0.05 inch.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

(5) Delamination is permitted if it is:

(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

C. Zone 1 - Edgeband Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the fiberglass or graphite fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the fiberglass or graphite fibers are permitted if they are:

(a) A maximum of one ply in depth.

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

(3) Dents:

(a) Dents that do not cause damage to the carbon glass fibers are permitted if they are:

1) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

**NOTE**: Use the limits for holes and punctures if there is carbon glass fiber damage or if the dent depth is more than 0.05 inch.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

(5) Delamination is permitted if it is:

(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

(6) Edge damage is permitted if it is:

(a) The maximum as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1 Details C and D if there is a fastener present.

(b) The maximum as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail E if there is not a fastener present.

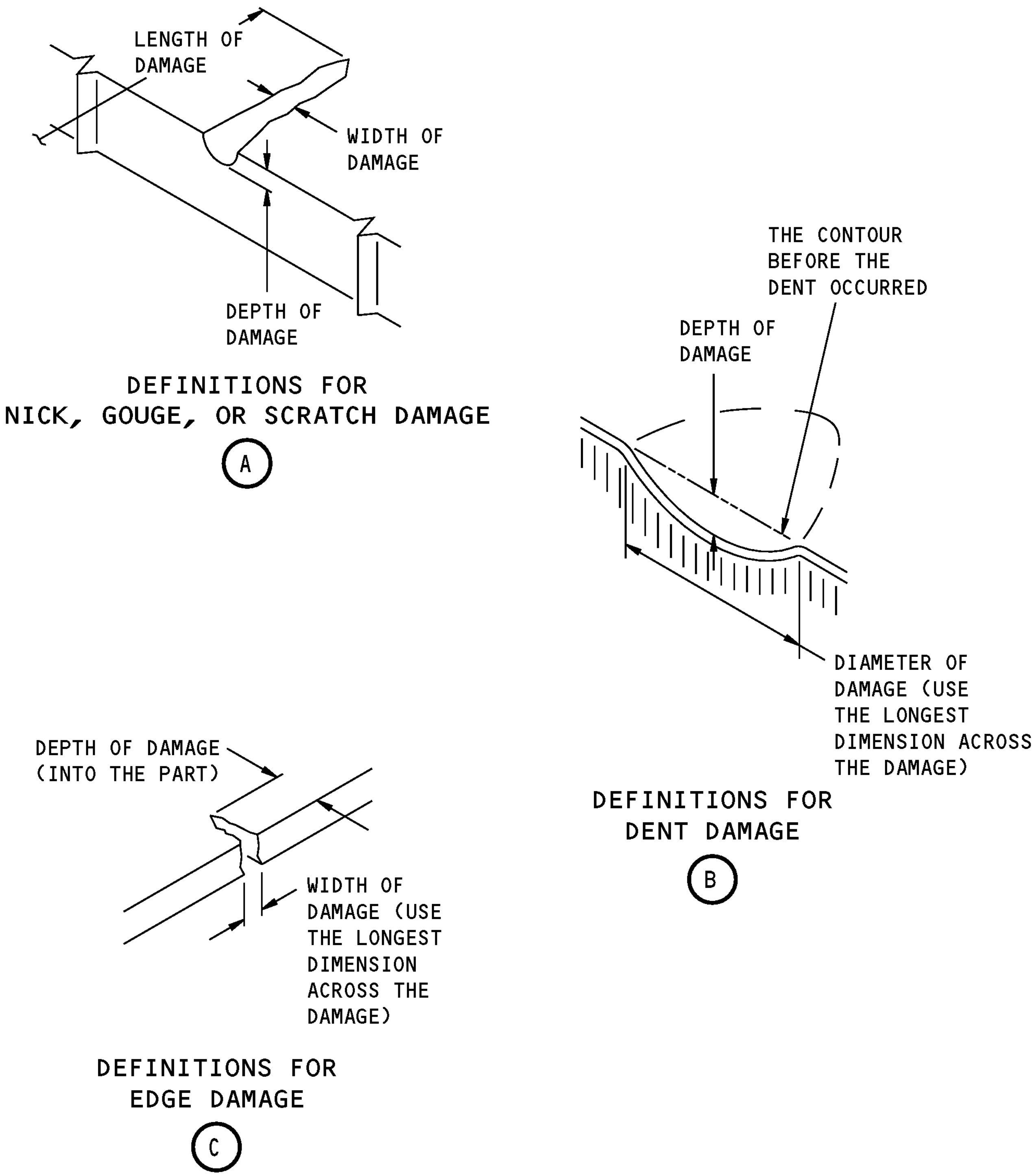
(c) Sealed as given in Paragrqaph 2.

(7) Edge Erosion is permitted if it is:

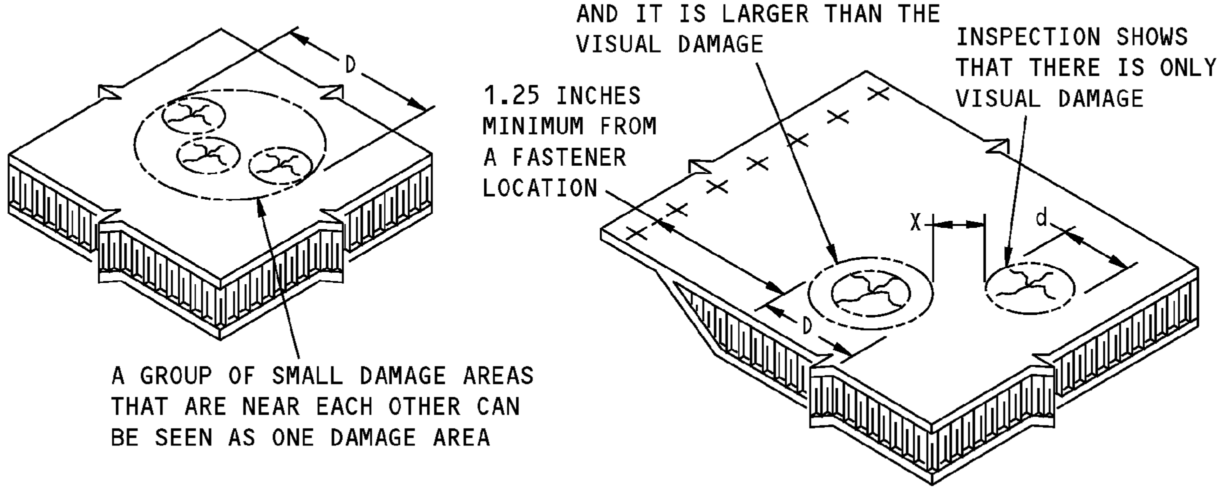
(a) The maximum as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail F.

D. Zone 2 - Honeycomb Core, Solid Laminate, and Edgeband Areas

(1) Damage is not permitted.




**Damage Definitions**
**Figure 105**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 2.0 INCHES.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

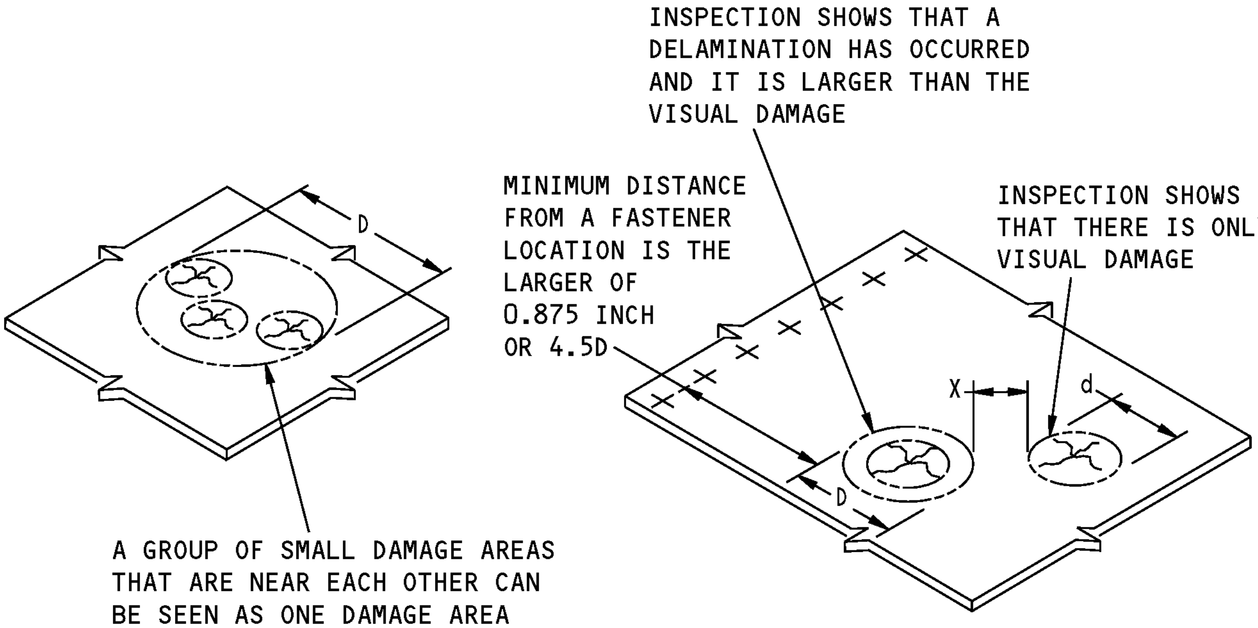
X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3D SPACING.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS (HONEYCOMB SANDWICH)

A

**Allowable Damage Limits**
**Figure 106 (Sheet 1 of 4)**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.25 INCH.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 4D SPACING.

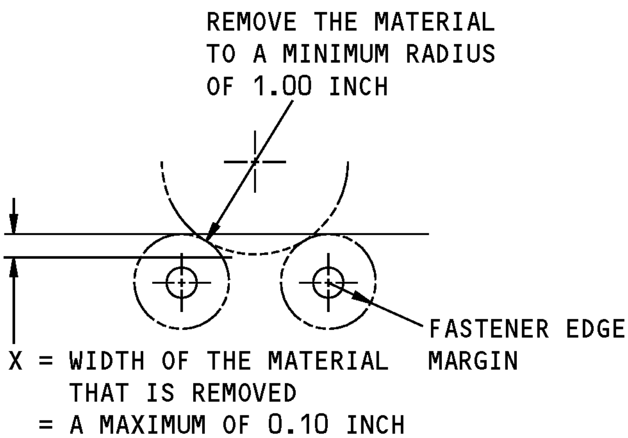
THE MINIMUM DISTANCE FROM AN EDGE IS 2.5D + 0.05 INCH.

THE MINIMUM DISTANCE FROM A DRAIN HOLE IS 4.5D.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS
(EDGEBANDS AND SOLID LAMINATES)

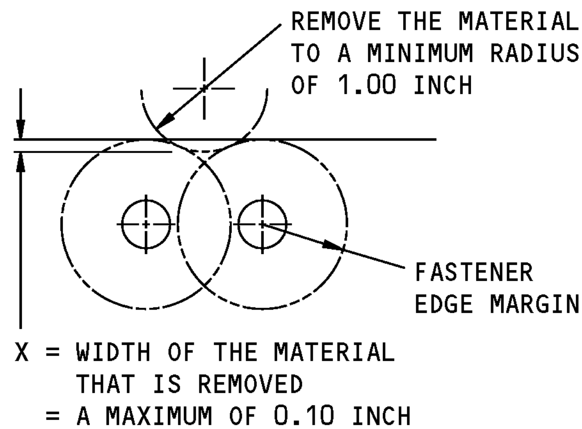
B

**Allowable Damage Limits**
**Figure 106 (Sheet 2 of 4)**



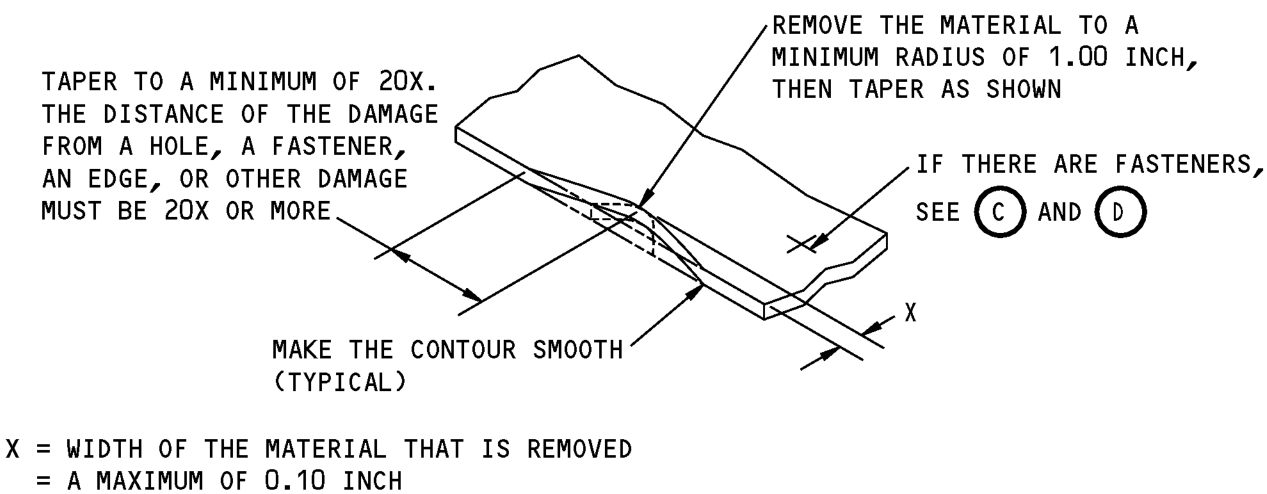

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

C




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

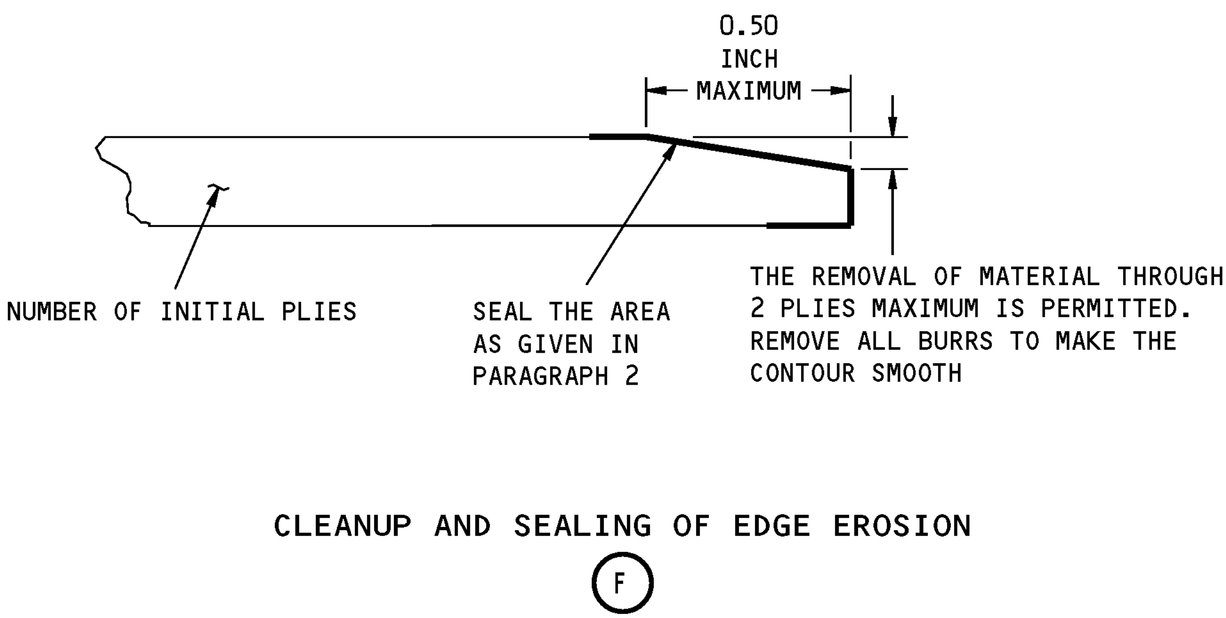
D




REMOVAL OF DAMAGED MATERIAL AT AN EDGEBAND

E

**Allowable Damage Limits**
**Figure 106 (Sheet 3 of 4)**




**Allowable Damage Limits**
**Figure 106 (Sheet 4 of 4)**

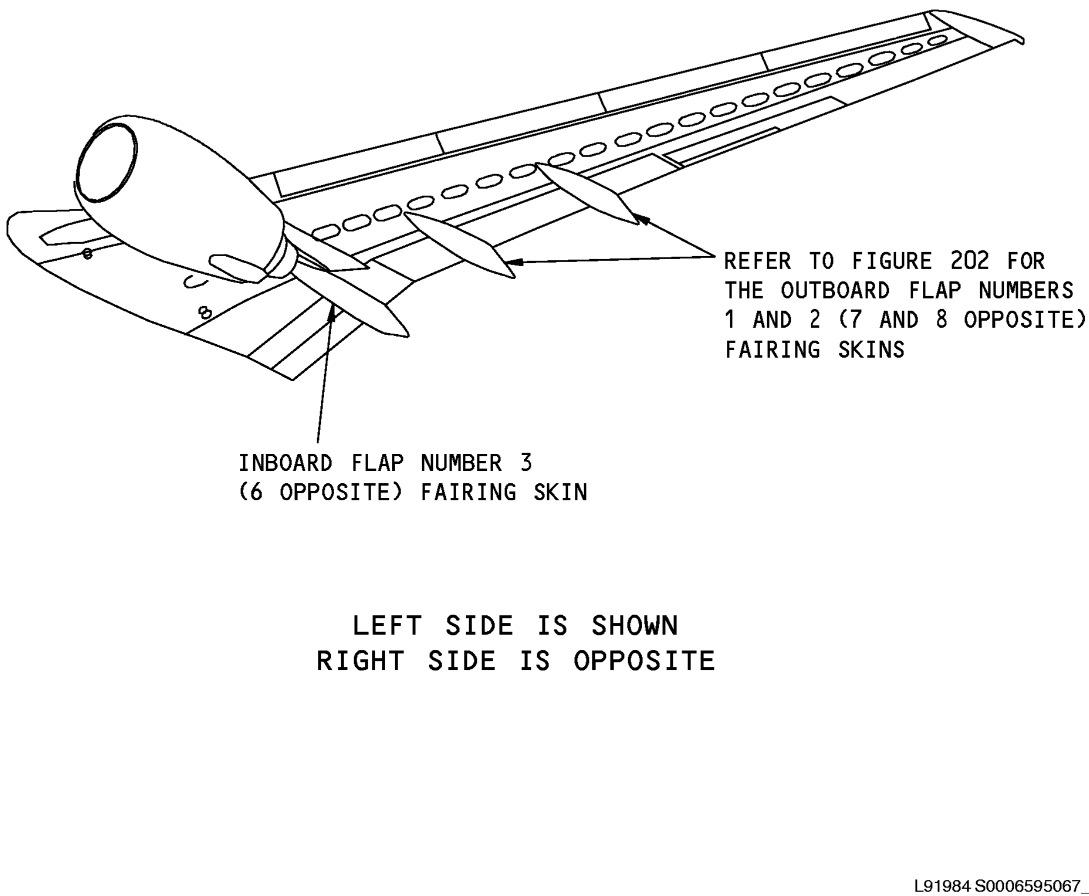
## REPAIR 1 - FLAP SUPPORT FAIRING SKIN, WING TRAILING EDGE FLAP

1. **Applicability**

   A. Repair 1 is applicable to the skin panels of the flap support fairing shown in Flap Support Fairing Location for the Wing Trailing Edge Flap, Figure 201/REPAIR 1.

   B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damgage 1 for the type and size of damage that is permitted.

   C. Contact The Boeing Company for a repair to the inboard flap numbers 3 and 6 fairing skins.




**Flap Support Fairing Location for the Wing Trailing Edge Flap**
**Figure 201**

2. **General**

   A. Repair 1 gives instructions for Category A repairs for the number 1 and 2 (7 and 8 opposite) outboard flap fairing skins. Refer to 51-00-06 to find the definitions of the different categories of repairs.

   B. Remove the skin as necessary to get access to the inner surface.

(1) Remove the fasteners. Refer to 51-40-02 for information on fastener removal.

(2) If the fastener hole is damaged, refer to 51-70-04 or 51-70-05 for a repair.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Nondestructive Inspection (NDI) procedure. Refer to NDT, Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

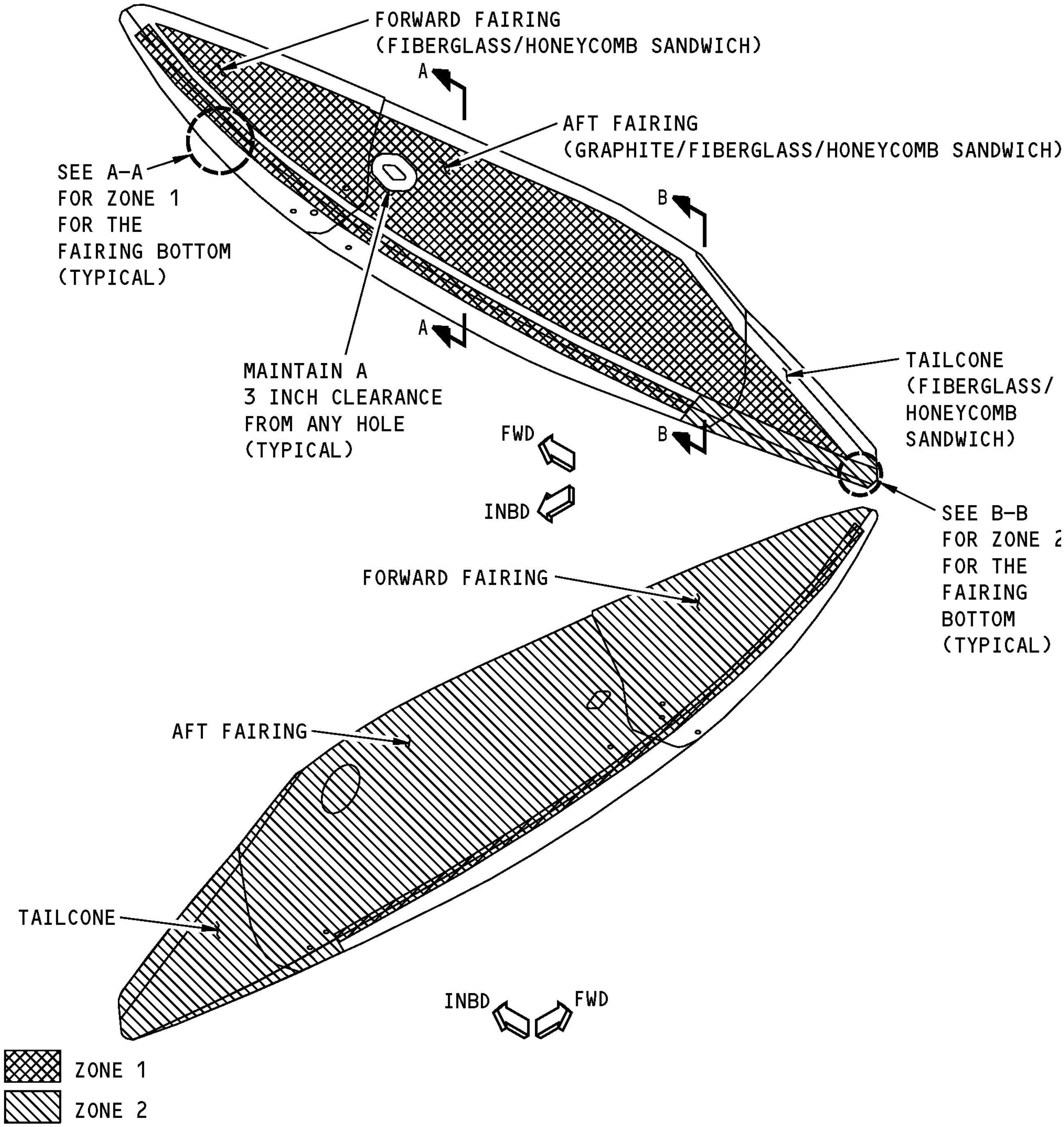
(1) Refer to Damage Definitions, Figure 203/REPAIR 1 for the definitions of the diameter and depth of damage.

(2) Refer to Definitions of the Facesheets, Figure 204/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

D. Do the repair as given in Paragraph 4./REPAIR 1

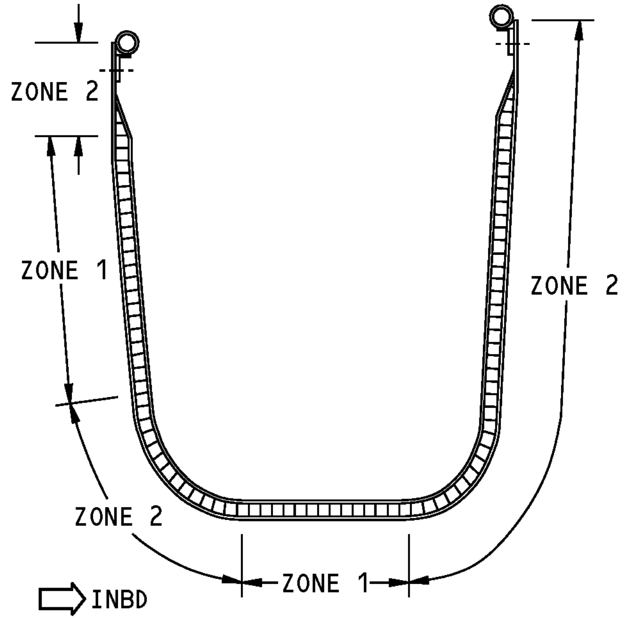
E. Install the skin, as necessary.

F. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

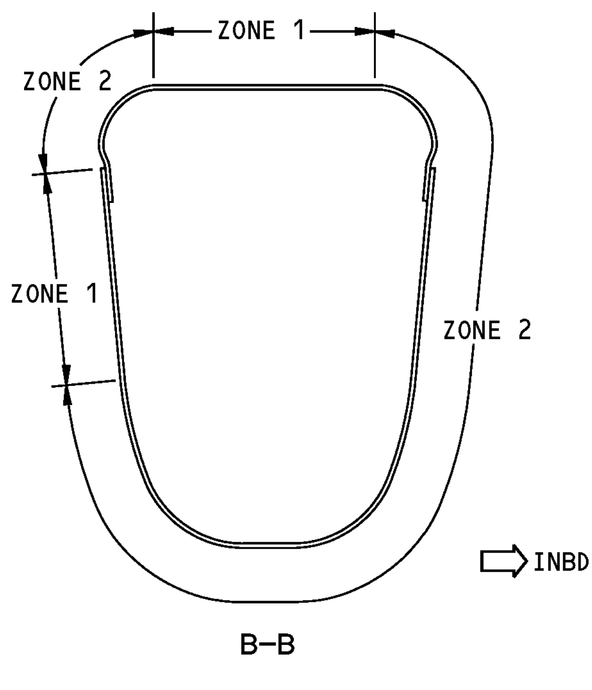



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
FLAP SUPPORT FAIRING NUMBER 1 IS SHOWN,
FLAP SUPPORT FAIRING NUMBER 2 IS SIMILAR

**Flap Support Fairing Numbers 1 and 2 For the Outboard Trailing Edge Flap**
**Figure 202 (Sheet 1 of 2)**




A-A

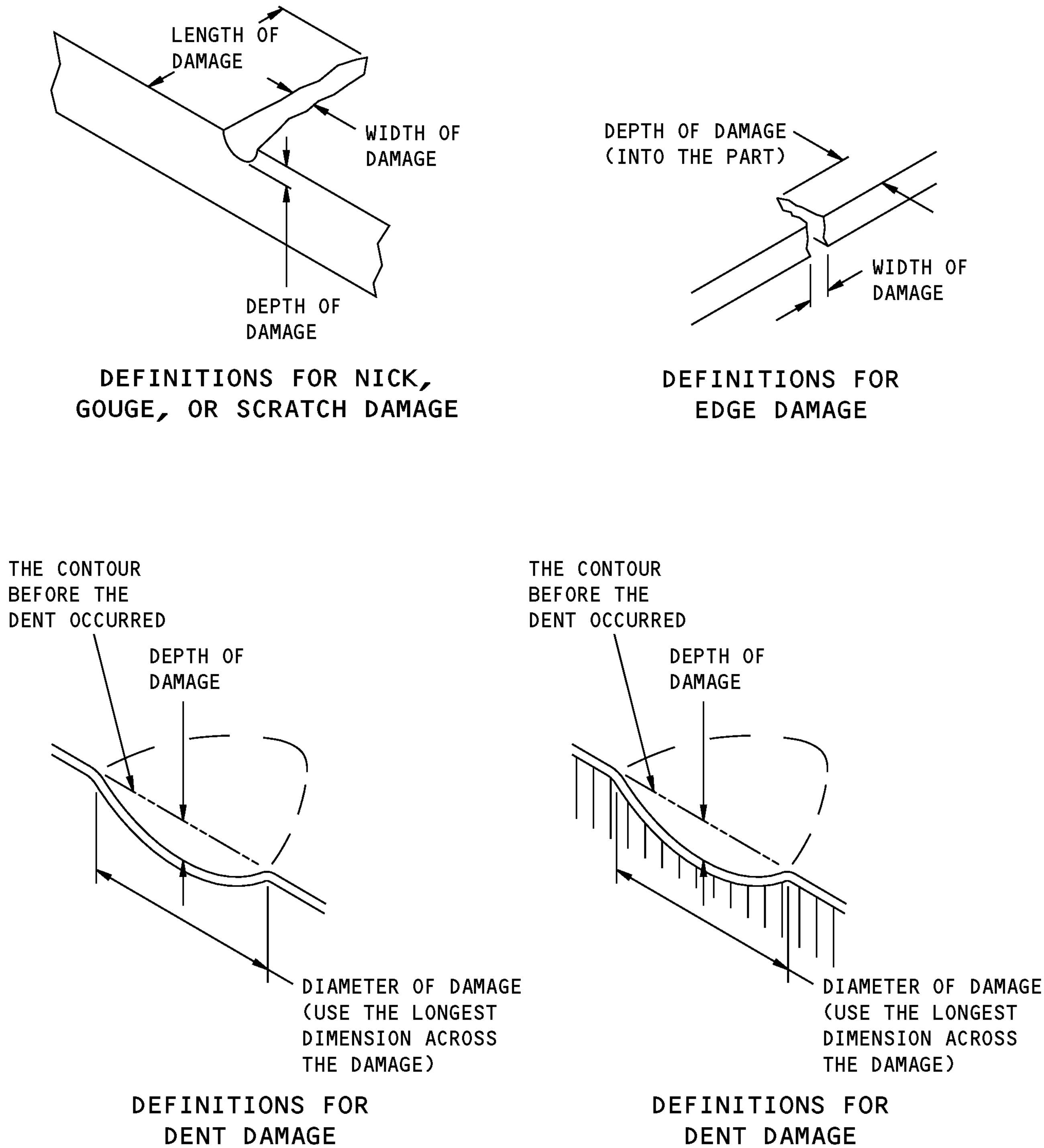



B-B

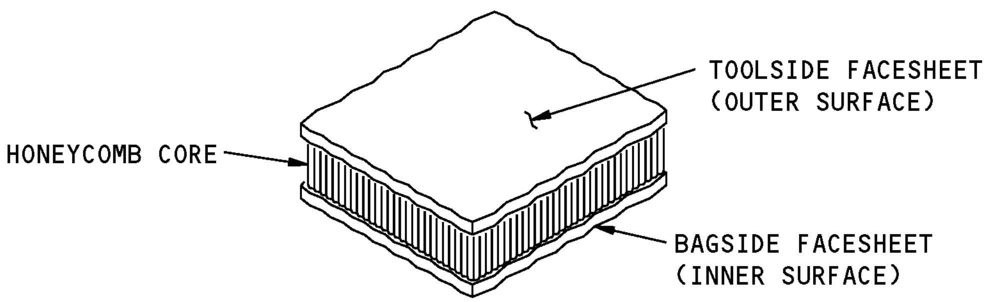
NOTE: REPAIRS PERMITTED IN ZONE 1 PROVIDED THAT:

REPAIRS PLIES DO NOT EXTEND INTO THE CORE RAMP AREA.
REPAIRS PLIES DO NOT EXTEND INTO CURVE PORTION OF THE CORNERS.

**Flap Support Fairing Numbers 1 and 2 For the Outboard Trailing Edge Flap**
**Figure 202 (Sheet 2 of 2)**




**Damage Definitions**
**Figure 203**




**Definitions of the Facesheets**
**Figure 204**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-05 | REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS |
| AMM 51-21-00 P/B 701 | INTERIOR AND EXTERIOR FINISHES - CLEANING/PAINTING |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

## 4. Repair Instructions

A. Refer to:

(1) Table 201/REPAIR 1 for Zone 1 honeycomb core areas

(2) Table 202/REPAIR 1 for the Zone 1 solid laminate areas.

B. Contact The Boeing Company for a Zone 2 repair or a repair made with:

(1) Room temperature wet layup materials

(2) Wet layup materials at 150 °F (66°C) cure.

C. Use the repair instructions that follow for a repair made with wet layup materials at 200 °F (121°C) cure.

(1) Repair the damage as given in 51-70-04 and do the steps that follow:

(a) Use the same number of structural repair plies as the number of initial plies that are removed. Make sure that you do not remove more than (3) initial plies.

(b) Add two extra structural repair plies and one non-structural fiberglass repair ply. Put one structural ply at ±45 degrees to the core ribbon direction and the other at 0 or 90 degrees. Use an optional ply direction for the non-structural ply.

D. Use the repair instructions that follow for a repair made with preimpregnated layup materials at 250 °F (177°C) cure.

(1) Repair the damage as given in 51-70-05 and do the steps that follow:

(a) Use the same number of structural repair plies as the number of initial plies that are removed.

(b) Add one extra non-structural fiberglass repair ply with an optional ply direction.

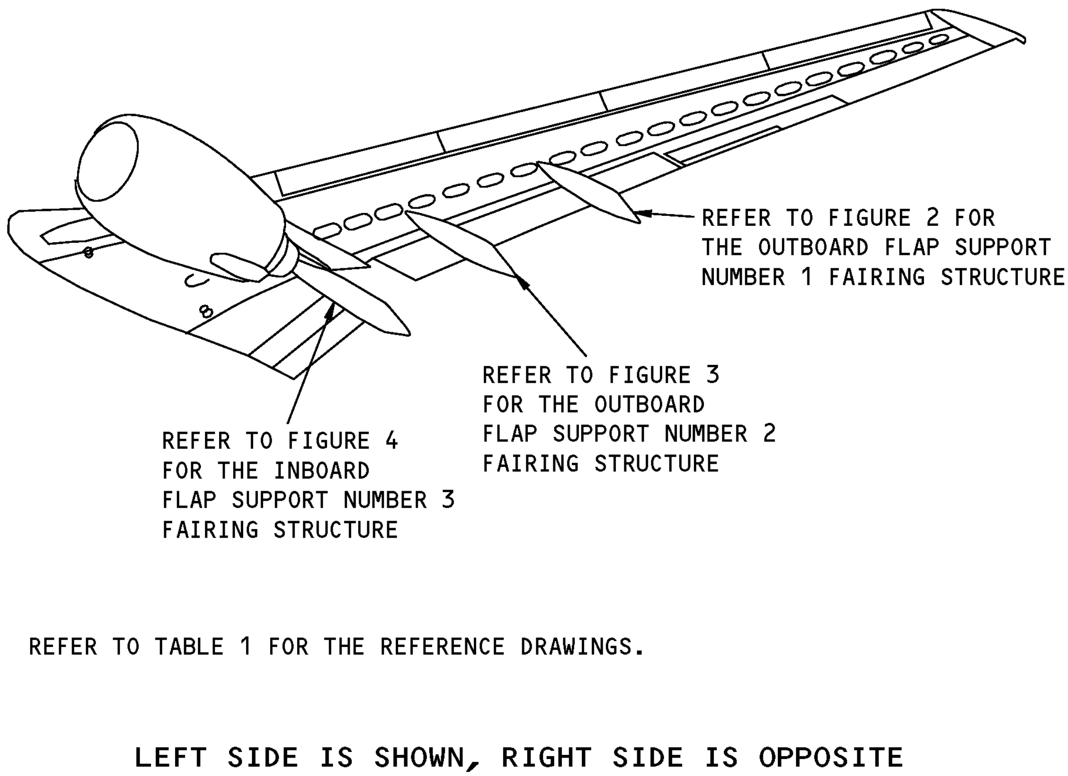
Table 201:

| ZONE 1 - REPAIR DATA FOR THE FLAP SUPPORT FAIRING SKIN 250°F (121°C) CURE, HONEYCOMB CORE PANEL AREA | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **ROOM TEMPERATURE** | **150F (66°C)** | **200°F (93°C)** | **250°F (121°C)** |
| REPAIR SIZE AND LIMITS | - Contact The Boeing Company for this temperature repair. | - Contact The Boeing Company for this temperature repair. | Damage that is a maximum of:<br>- 2.0 inches in diameter<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- panel edges<br>- fastener holes | There are no limits on the dimension of the repair. |
| REPAIR PROCEDURES | —— | ——— | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-05 and Paragraph 4.D |

Table 202:

| ZONE 1 - REPAIR DATA FOR THE FLAP SUPPORT FAIRING SKIN 250°F (121°C) CURE, SOLID LAMINATE PANEL AREAS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **ROOM TEMPERATURE** | **150F (66°C)** | **200°F (93°C)** | **250°F (121°C)** |
| REPAIR SIZE AND LIMITS | - Contact The Boeing Company for this temperature repair. | - Contact The Boeing Company for this temperature repair. | - Contact The Boeing Company for this temperature repair. | There are no limits on the dimension of the repair. |
| REPAIR PROCEDURES | —— | ——— | ——— | SRM 51-70-05 and Paragraph 4.D |

## IDENTIFICATION 1 - FLAP SUPPORT FAIRINGS STRUCTURE

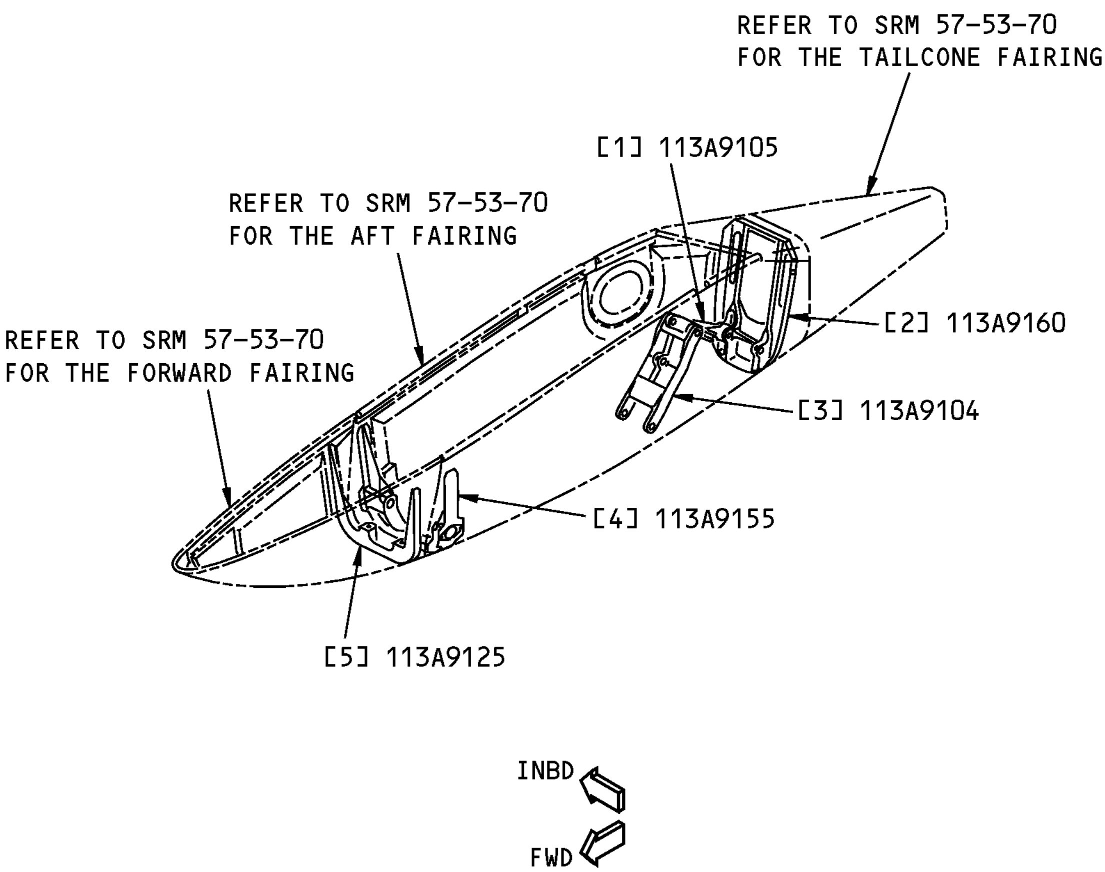



**Flap Support Fairing Location for the Wing Trailing Edge Flap**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 001A0101 | Final Assembly - Product Collector |
| 113A0001 | Movable T.E. Functional Product Collector |
| 113A9100 | Fairing Installation - Flap Support No. 1 and 8, Outbd T.E. Flap |
| 113A9104 | Attach Beam Assembly - Aft Fairing, Flap Support No. 1 |
| 113A9105 | Adjust Link Assembly - Aft Fairing, Flap Support No. 1 |
| 113A9110 | FWD Fairing Assembly - Flap Support No. 1, Outbd T.E. Flap |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A9150 | Aft Fairing Assembly - Flap Support No. 1, Outbd T.E. Flap |
| 113A9155 | Pivot Frame Assembly - Aft Fairing, Flap Support No. 1 |
| 113A9160 | Aft Bulkhead Assembly - Aft Fairing, Flap Support No. 1 |
| 113A9200 | Fairing Installation - Flap Support No. 2 and 7, Outbd T.E. Flap |
| 113A9204 | Attach Beam Assembly - Aft Fairing, Flap Support No. 2 |
| 113A9216 | Attach Link Assembly - Aft Fairing, Flap Support No. 2 |
| 113A9205 | Adjust Link Assembly - Aft Fairing, Flap Support No. 2 |
| 113A9210 | FWD Fairing Assembly - Flap Support No. 2, Outbd T.E. Flap |
| 113A9250 | Aft Fairing Assembly - Flap Support No. 2, Outbd T.E. Flap |
| 113A9255 | Pivot Frame Assembly - Aft Fairing, Flap Support No. 2 |
| 113A9260 | Aft Bulkhead Assembly - Aft Fairing, Flap Support No. 2 |
| 113A9300 | Fairing Installation - Flap Support No. 3 and 6, Outbd T.E. Flap |
| 113A9304 | Attach Beam Assembly - Aft Fairing, Flap Support No. 3 |
| 113A9307 | Adjust Link Assembly - Aft Fairing, Flap Support No. 3 |
| 113A9310 | FWD Fairing Assembly - Flap Support No. 3, Outbd T.E. Flap |
| 113A9350 | Aft Fairing Assembly - Flap Support No. 3, Outbd T.E. Flap |
| 113A9355 | Pivot Frame Assembly - Aft Fairing, Flap Support No. 3 |
| 113A9358 | Mid Frame Assembly - Aft Fairing, Flap Support No. 3 |
| 113A9360 | Aft Bulkhead Assembly - Aft Fairing, Flap Support No. 3 |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

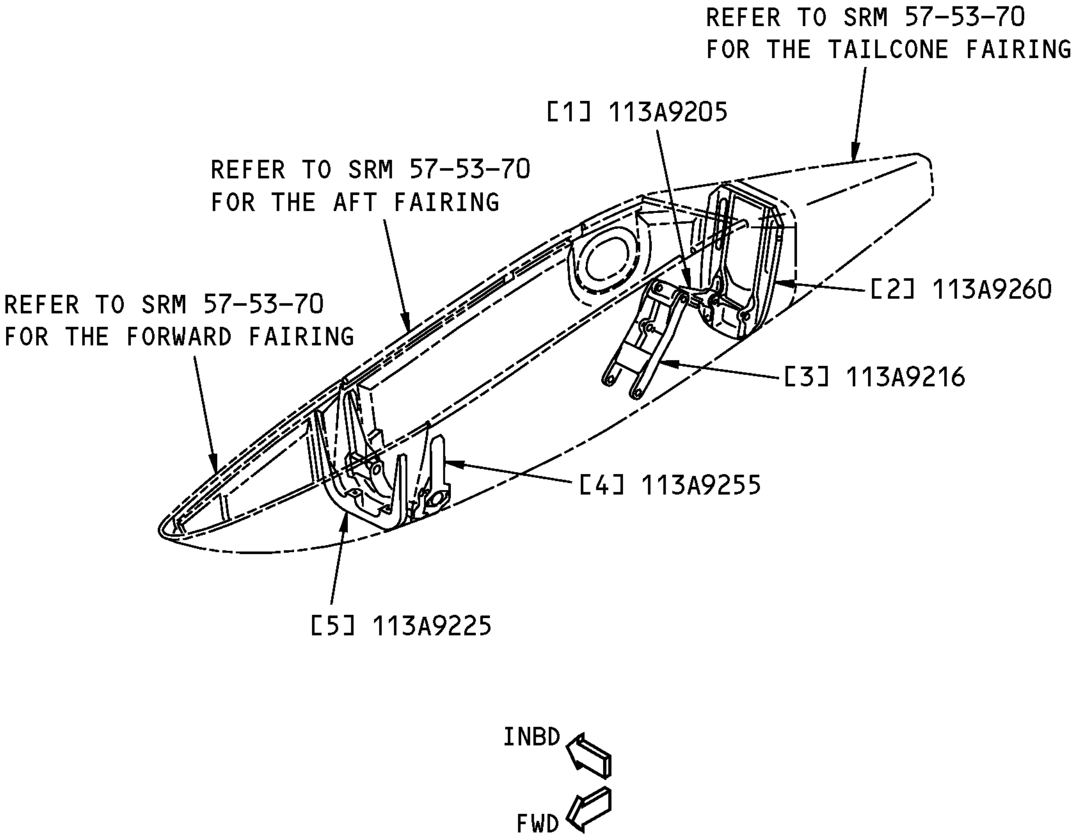
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Flap Support Fairing Structure for the Number 1 Outboard Trailing Edge Flap**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Adjust Link | 1.9 (48.3) | 7050-T7451 plate as given in AMS 4050 | |
| [2] | Bulkhead | 2.25 (57.2) | 7050-T7451 plate as given in AMS 4050 | |
| [3] | Attach Beam | 2.76 (70.1) | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Pivot Frame | 3.8 (96.5) | 7050-T7451 plate as given in AMS 4050 | |
| [5] | Frame | 1.5 (38.1) | 7050-T7451 plate as given in AMS 4050 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

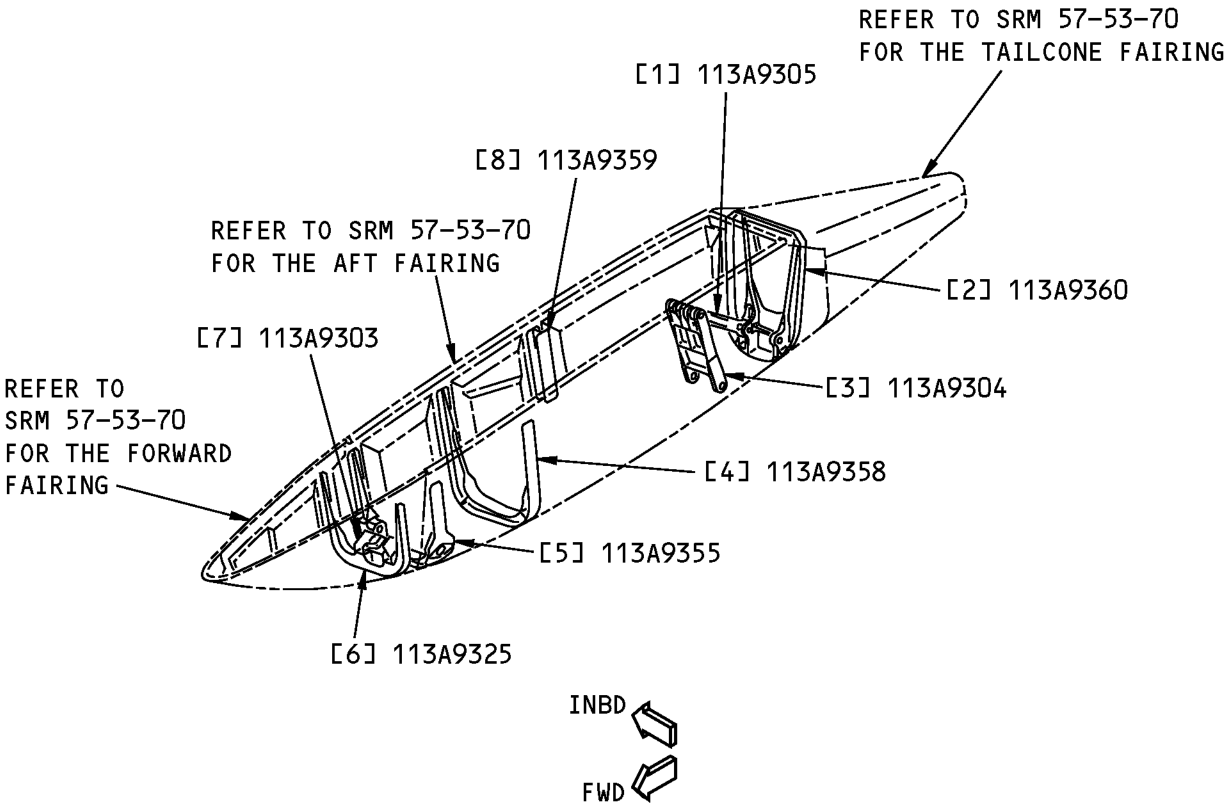
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Flap Support Fairing Structure for the Number 2 Outboard Trailing Edge Flap**
**Figure 3**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Adjust Link | 1.9 (48.3) | 7050-T7451 plate as given in AMS 4050 | |
| [2] | Bulkhead | 2.25 (57.2) | 7050-T7451 plate as given in AMS 4050 | |
| [3] | Attach Beam | 2.86 (72.6) | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Pivot Frame | 3.8 (96.5) | 7050-T7451 plate as given in AMS 4050 | |
| [5] | Frame | 1.5 (38.1) | 7050-T7451 plate as given in AMS 4050 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Flap Support Fairing Structure for the Number 3 Inboard Trailing Edge Flap**
**Figure 4**

**Table 4**:

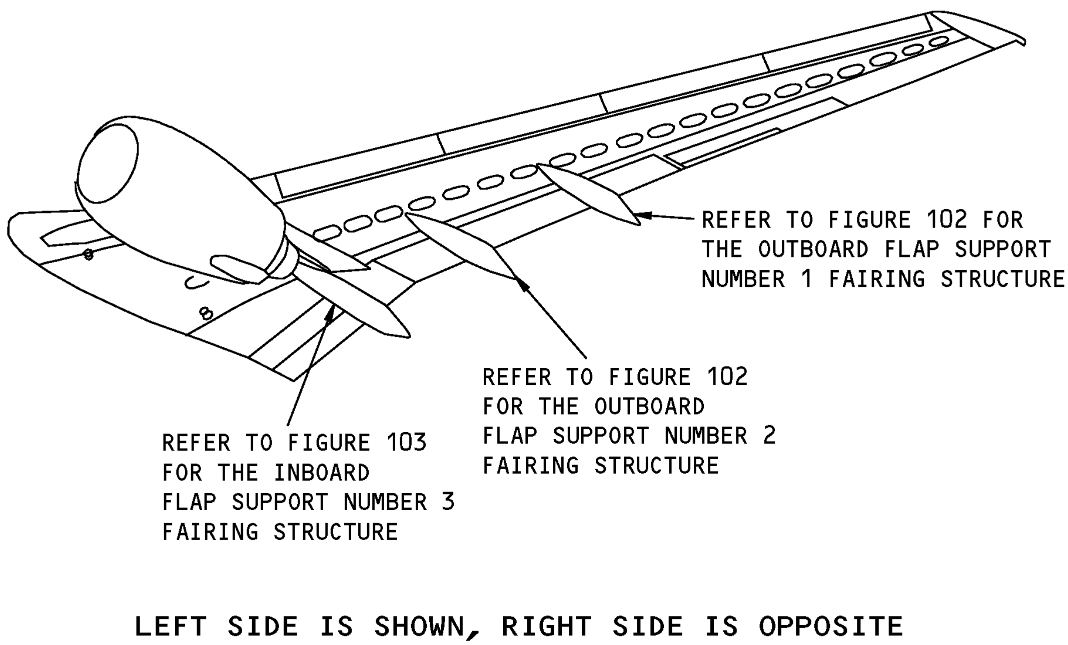
| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Adjust Link | 1.9 (48.3) | 7050-T7451 plate as given in AMS 4050 | |
| [2] | Bulkhead | 2.25 (57.2) | 7050-T7451 plate as given in AMS 4050 | |
| [3] | Attach Beam | 1.64 (41.7) | 7050-T7451 plate as given in AMS 4050 | |
| [4] | Mid Frame | 1.8 (45.7) | 7050-T7451 plate as given in AMS 4050 | |
| [5] | Pivot Frame | 3.8 (96.5) | 7050-T7451 plate as given in AMS 4050 | |
| [6] | Frame | 1.5 (38.1) | 7050-T7451 plate as given in AMS 4050 | |
| [7] | Attach Fitting | 1.96 (49.8) | 7050-T7451 plate as given in AMS 4050 | |
| [8] | Stub Beam | 1.8 (45.7) | 7050-T7451 plate as given in AMS 4050 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

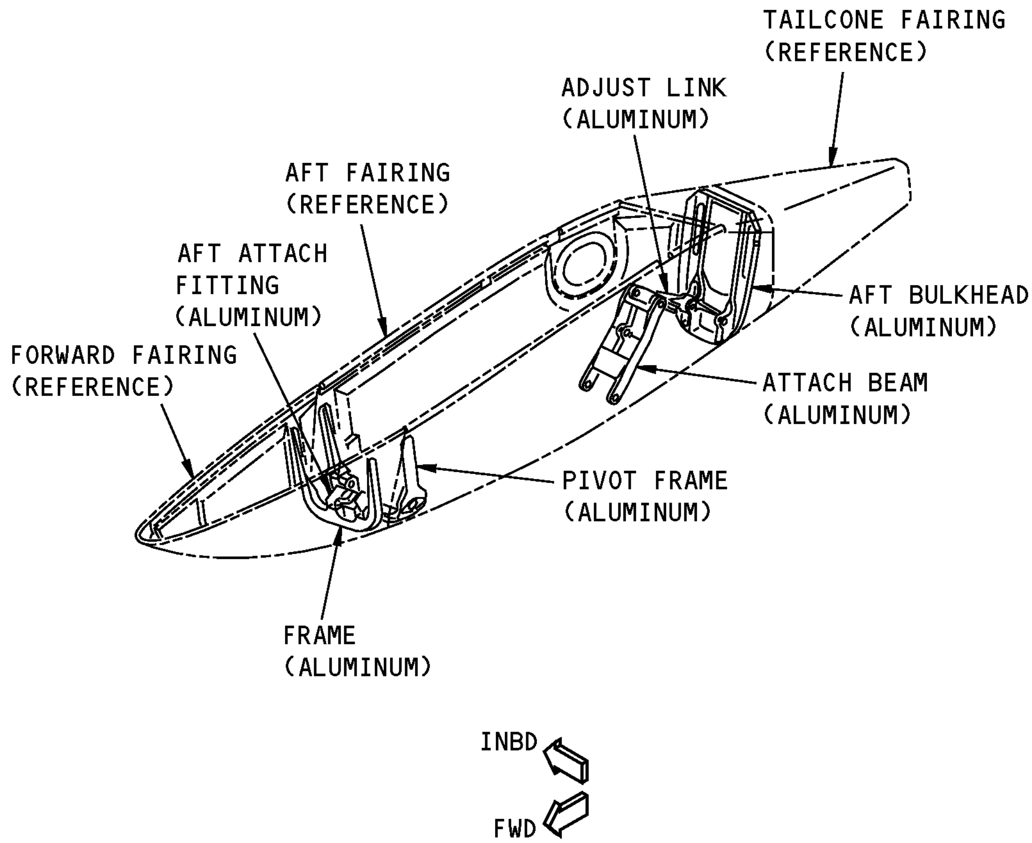
## ALLOWABLE DAMAGE 1 - WING TRAILING EDGE FLAP TRACK FAIRING SUPPORT STRUCTURE

1. **Applicability**

   A. This subject gives the allowable damage limits for the wing trailing edge flap track fairing support structure shown in Flap Support Fairing Location for the Wing Trailing Edge Flap, Figure 101/ALLOWABLE DAMAGE 1.



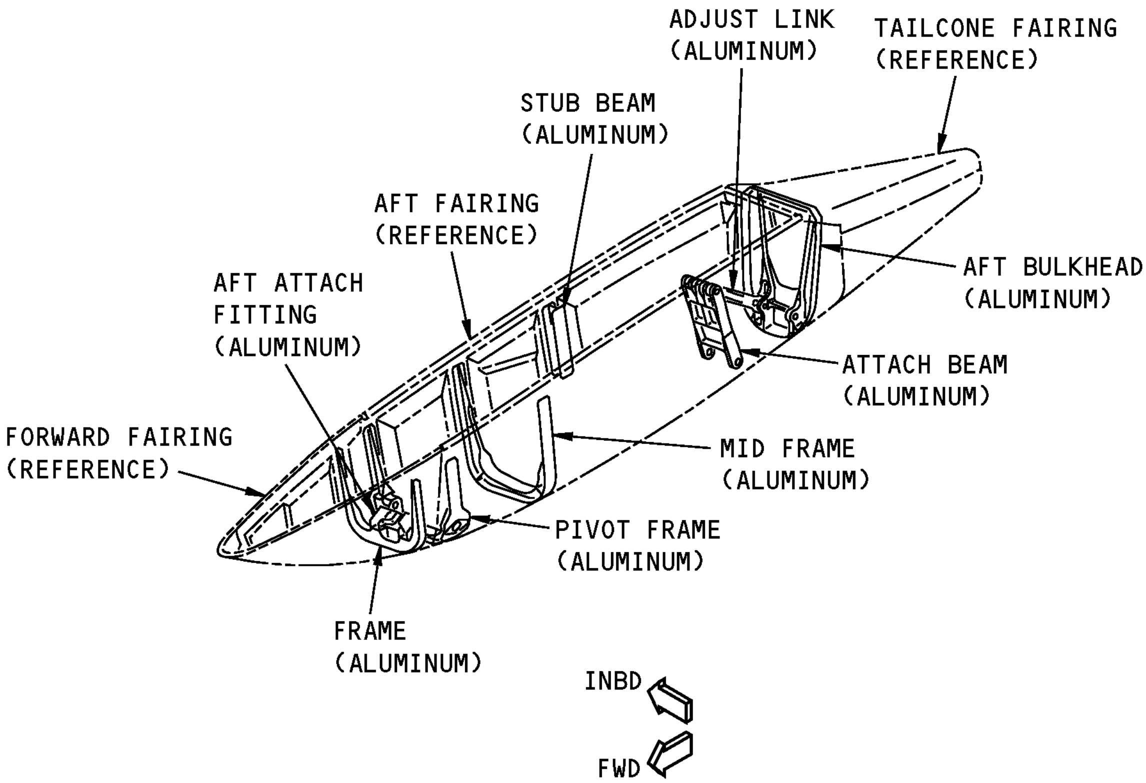

**Flap Support Fairing Location for the Wing Trailing Edge Flap**
**Figure 101**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

FAIRING NUMBER 1 IS SHOWN, FAIRING NUMBER 2 IS ALMOST THE SAME

**Flap Support Fairing Structures Number 1 and Number 2 for the Outboard Trailing Edge Flap**
**Figure 102**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Flap Support Fairing Structure Number 3 for the Inboard Trailing Edge Flap**
**Figure 103**

**2. General**

A. Do the steps that follow if you have damage to the aluminum parts:

(1) Remove the damage as necessary.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) After you remove the damage, do the procedures that follow:

(a) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(b) Apply a layer of BMS 10-11, Type I primer to the surfaces of the aluminum parts. Refer to SOPM 20-41-02.

**3. References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-40-08 | COUNTERSINKING |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

**4. Allowable Damage Limits**

A. Frames, Bulkheads, Adjust Links, and Beams

(1) Cracks:

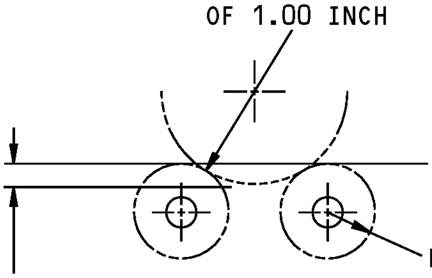
(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and F.

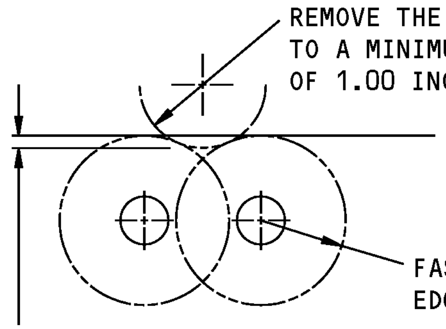
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



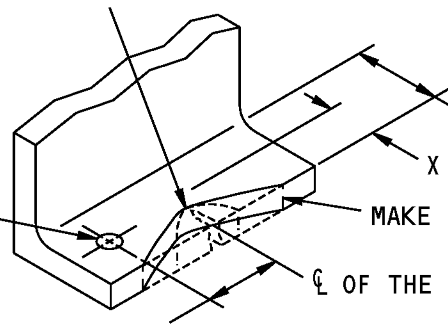

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

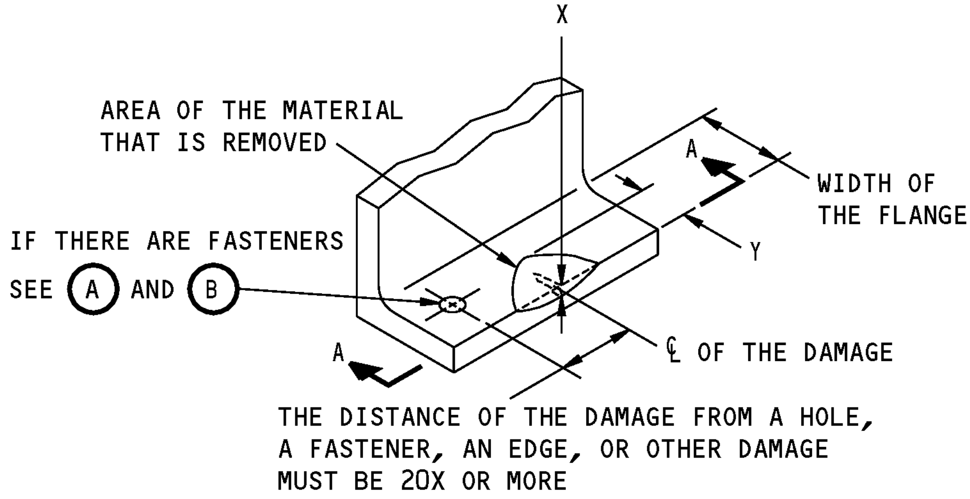
B




REMOVAL OF DAMAGED MATERIAL AT AN EDGE

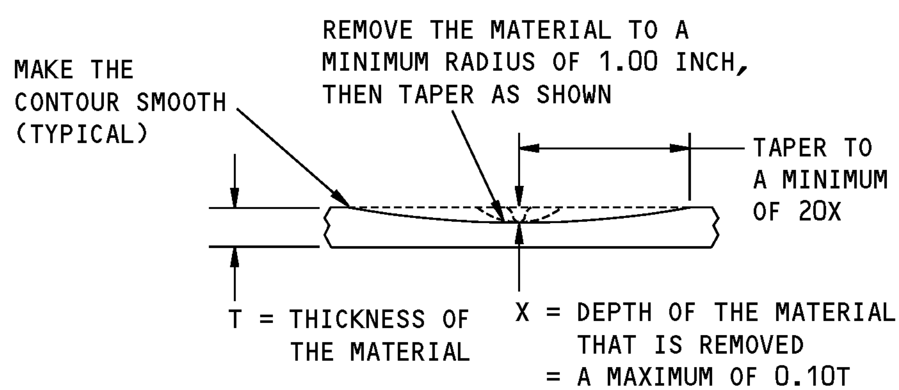
C

**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**



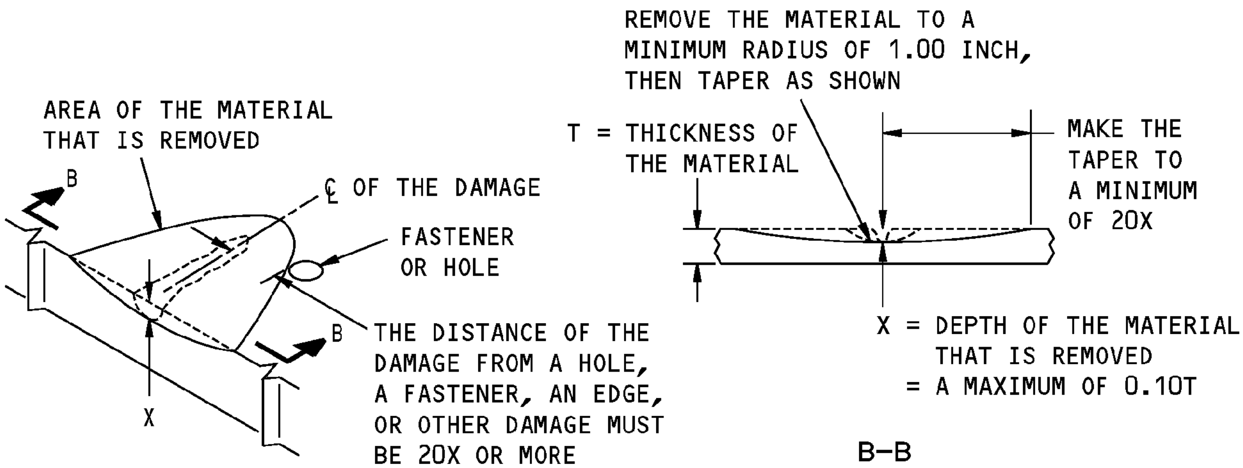

Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE
D



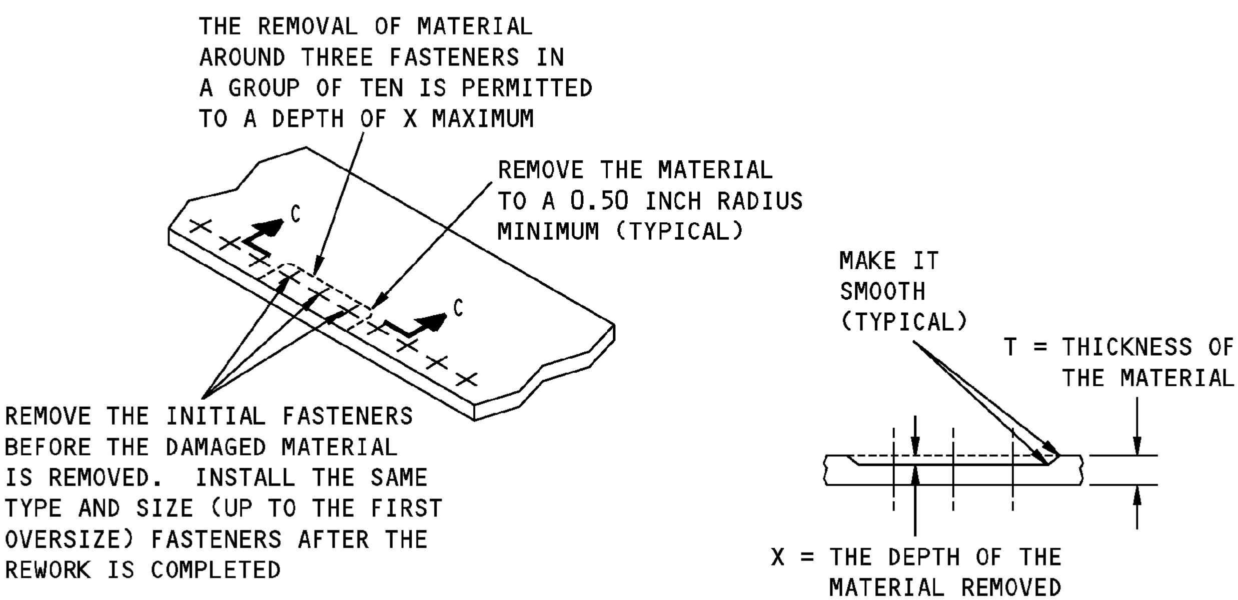

A-A

**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

E




REMOVAL OF CORROSION
AROUND THE FASTENERS

F

**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**

## REPAIR 1 - WING TRAILING EDGE FLAP TRACK FAIRING SUPPORT STRUCTURE

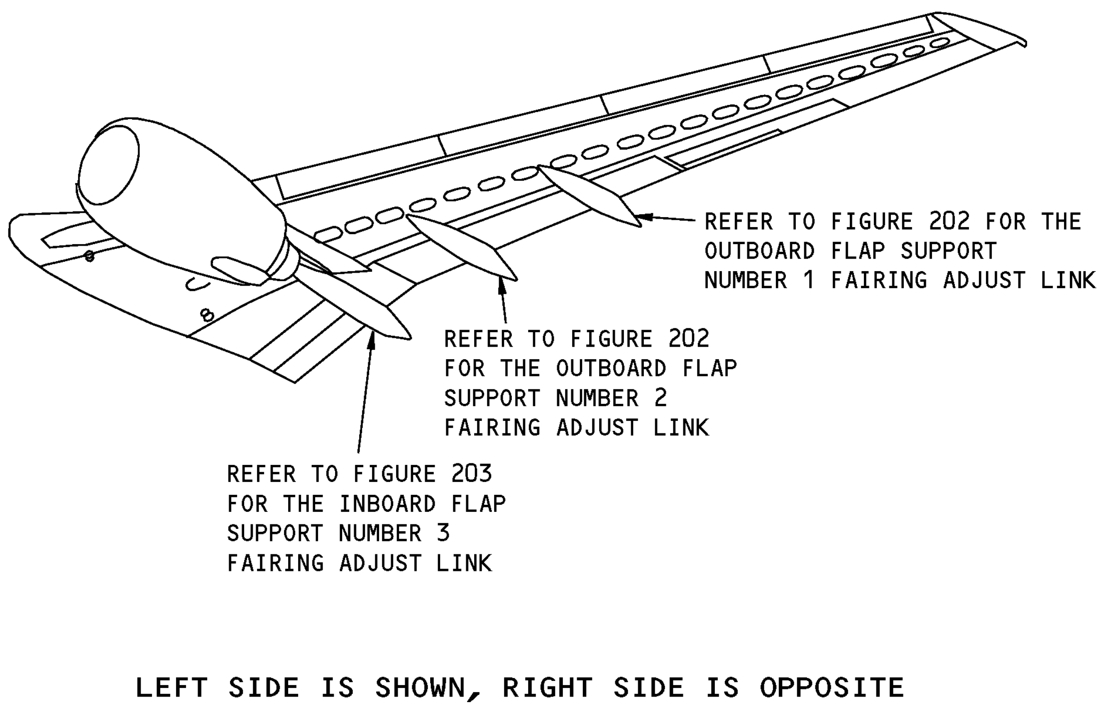
1. **Applicability**

   A. Repair 1 is applicable to damage to the flap support fairing adjust link as shown in Figure 201.

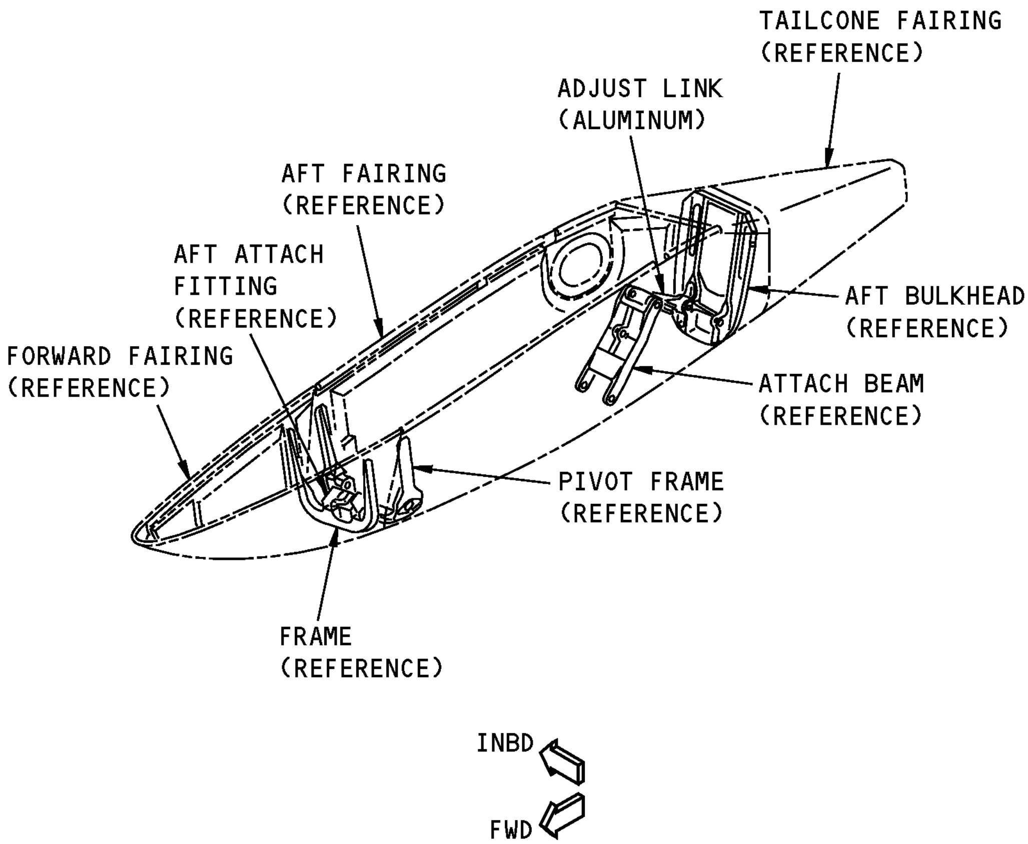
   B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the size and type of damage that is permitted.

2. **General**

   A. Repair 1 is a Category A repair. Refer to 51-00-06, GENERALto find the different categories of repairs.



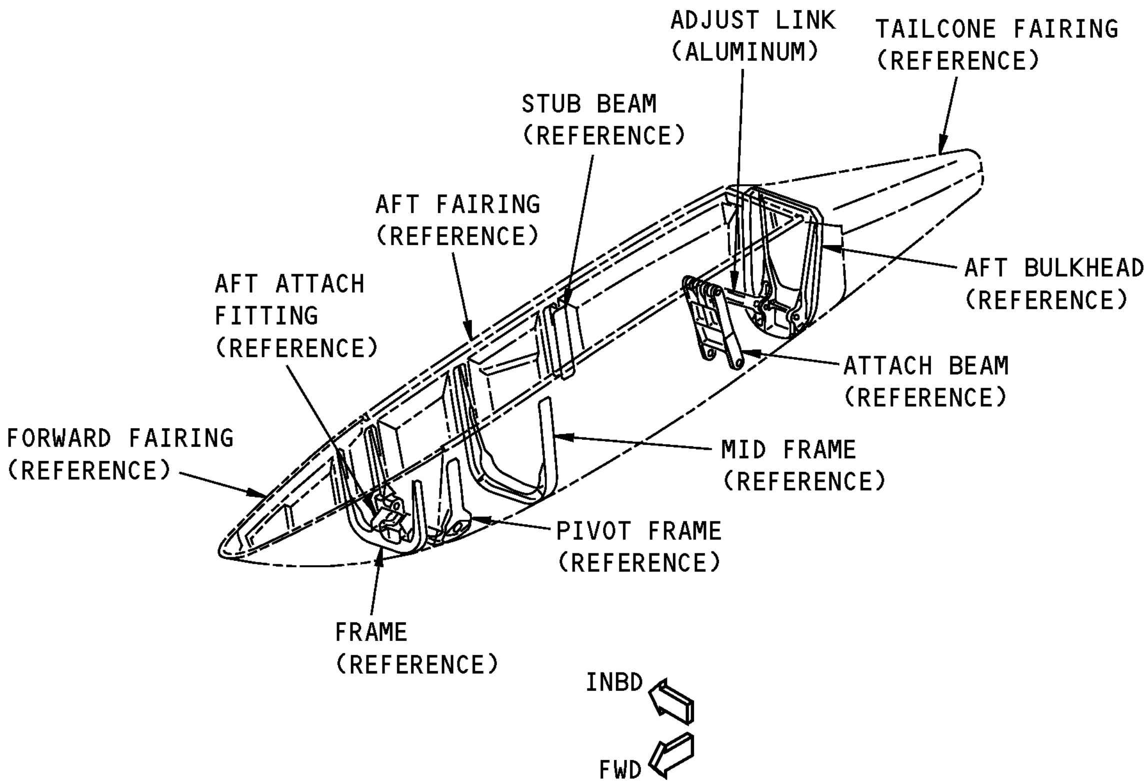

**Flap Support Fairing Location for the Wing Trailing Edge Flap**
**Figure 201**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

FAIRING NUMBER 1 IS SHOWN, FAIRING NUMBER 2 IS ALMOST THE SAME

**Adjust Link For the Flap Support Fairing Structure Numbers 1 and 2**
**Figure 202**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Adjust Link For the Flap Support Fairing Structure Number 3**
**Figure 203**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06, GENERAL | Structural Repair Definitions |
| 51-20-05, GENERAL | Repair Sealing |
| AMM 27-51-18 P/B 401 | INBOARD FLAP SUPPORT FAIRING - REMOVAL/INSTALLATION |
| AMM 27-51-18 P/B 501 | INBOARD FLAP SUPPORT FAIRING - ADJUSTMENT/TEST |
| AMM 27-51-28 P/B 401 | OUTBOARD FLAP SUPPORT FAIRINGS - REMOVAL/INSTALLATION |
| AMM 27-51-28 P/B 501 | OUTBOARD FLAP SUPPORT FAIRINGS - ADJUSTMENT/TEST |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-42-05 | Bright Cadmium Plating |

## 4. Repair Instructions

A. Remove the adjust link. Refer to AMM 27-51-18 and 27-51-28. Remove all necessary bolts, retainer plates, and bearings from the lug.

B. Make sure that you machine the lower lug bore between 1.080 (27.4 mm) and 1.085 (27.56 mm) inch diameter. Refer to Figure 204.

(1) Make the surface finish to all cut surfaces 63 microinches Ra or smoother.

C. Remove material from the lug thickness on the side opposite the shimmed side. Make sure that the final lug thickness is between 0.47 inch (11.94 mm) and 0.48 inch (12.19 mm). Refer to Figure 204. Blend the machined lug surface into the adjacent flange.

(1) Make the surface finish to all cut surfaces 125 microinches Ra or smoother.

D. Dye penetrant inspect the lug bore as given in SOPM 20-20-02.

E. Apply one coat of BMS 10-79, Type III primer to the reworked locations other than the bore.

F. Apply one coat of BMS 10-60, Type I enamel to the reworked locations other than the bore.

G. Use bushing part number 113A9120-1 or fabricate a repair bushing as shown in Figure 204 from 17-4PH or 15-5PH CRES, condition A and do as follows:

**NOTE**: The four bolt holes shown in Figure 204 may be drilled after the bushing has been installed on the lug. It is permitted to use a 1/64 inch oversize fastener.

(1) If a repair bushing is fabricated, then heat treat it to 180 to 200 KSI and provide a CAD plate as given in SOPM 20-42-05, type II, class 2. Continue to complete all other part number 113A9120-1 requirements.

(2) Use the shrink fit method to install the bushing as given in BAC5435. Install the bushing with BMS 5-95 sealant in the lug bore. Refer to 51-20-05, GENERAL. Make sure you provide a fillet seal around the bushing edges.

(3) If necessary, obtain a new KSC278607BZ bearing as shown in Figure 204 and install wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

H. If the gap between the lug and the retainer plate is more than 0.004 inch (0.10 mm) as shown in Figure 204 then install a new part number 69B14136-3 shim to get a 0.001 (0.03 mm) to 0.004 inch (0.10 mm) gap prior to bolt installation and do as follows:

**NOTE**: As an alternative, fabricate the shim from BAC1534-15C or machine the shim from 301 CRES.

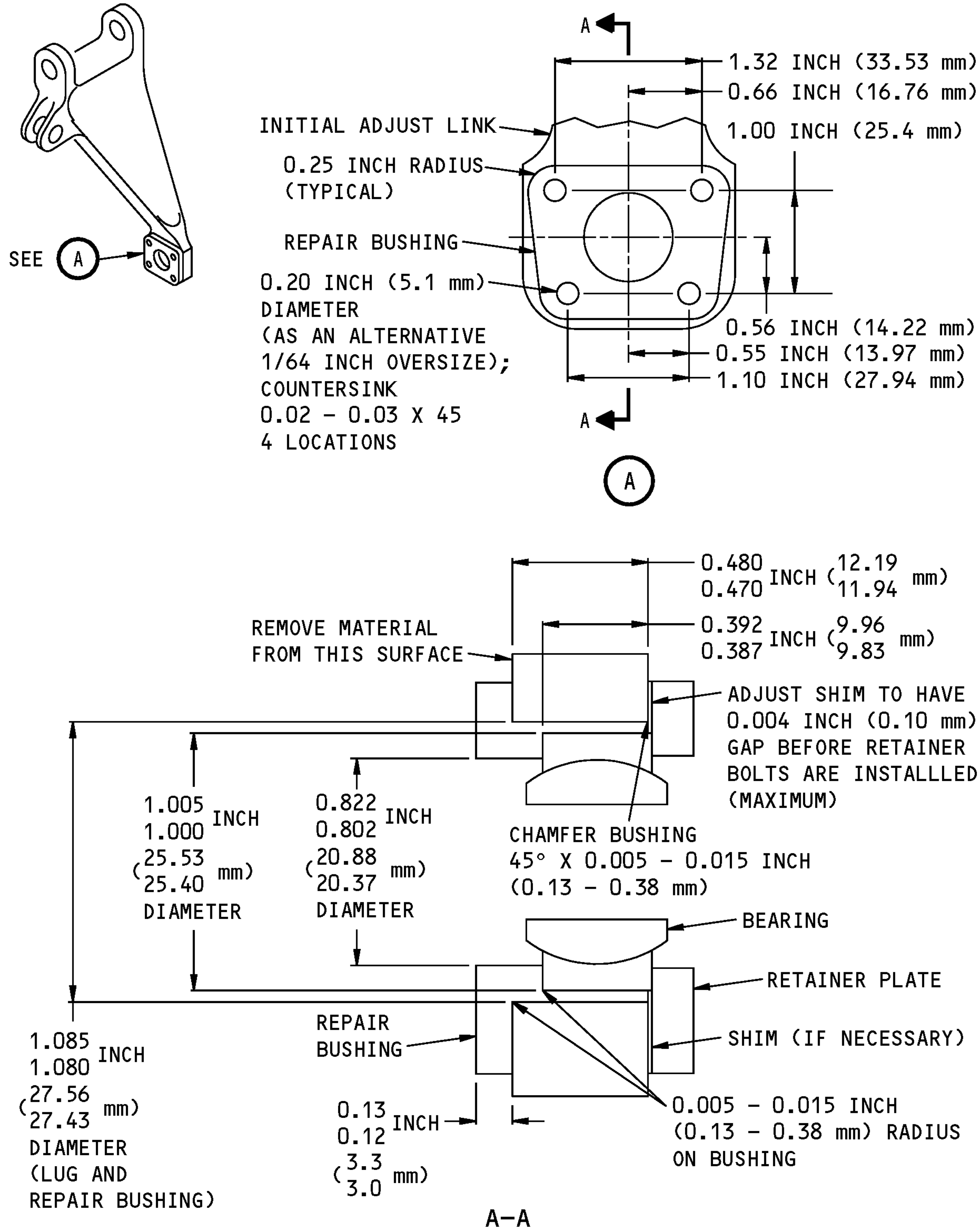
(1) Install the shim wet with BMS 5-95 sealant as given in 51-20-05, GENERAL.

(2) Apply one layer of BMS 10-11, Type I primer to the shim as given in SOPM 20-41-02.

I. Install the retainer with (4) BACB30NM3HK7 bolts and the BACW10DS3S washers. It is permitted to use 1/64 inch oversize for the bolt diameter.

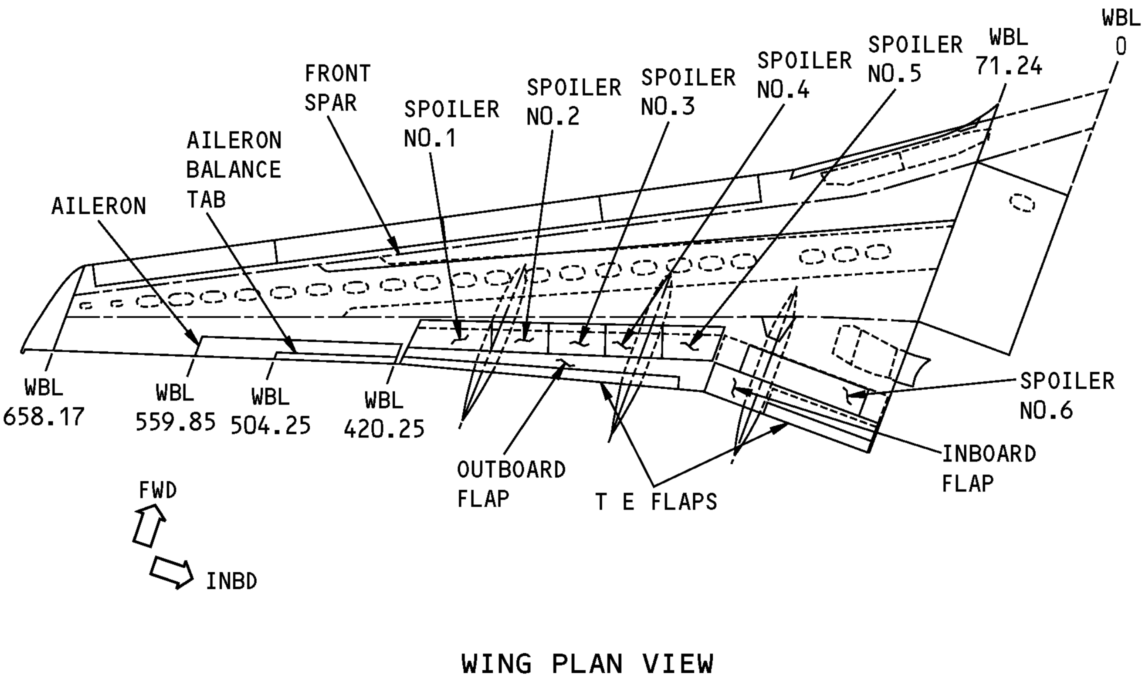
J. Lock wire the bolts together into two groups of two bolts. Use the double twist method as given in SOPM 20-50-02.

K. Reinstall the adjust link on the fairing as given in AMM PAGEBLOCK 27-51-18/401, AMM PAGEBLOCK 27-51-18/501, or AMM PAGEBLOCK 27-51-28/401, AMM PAGEBLOCK 27-51-28/501.




**Flap Support Fairing Adjust Link Repair**
**Figure 204**

## GENERAL - AILERON DIAGRAM




NOTES

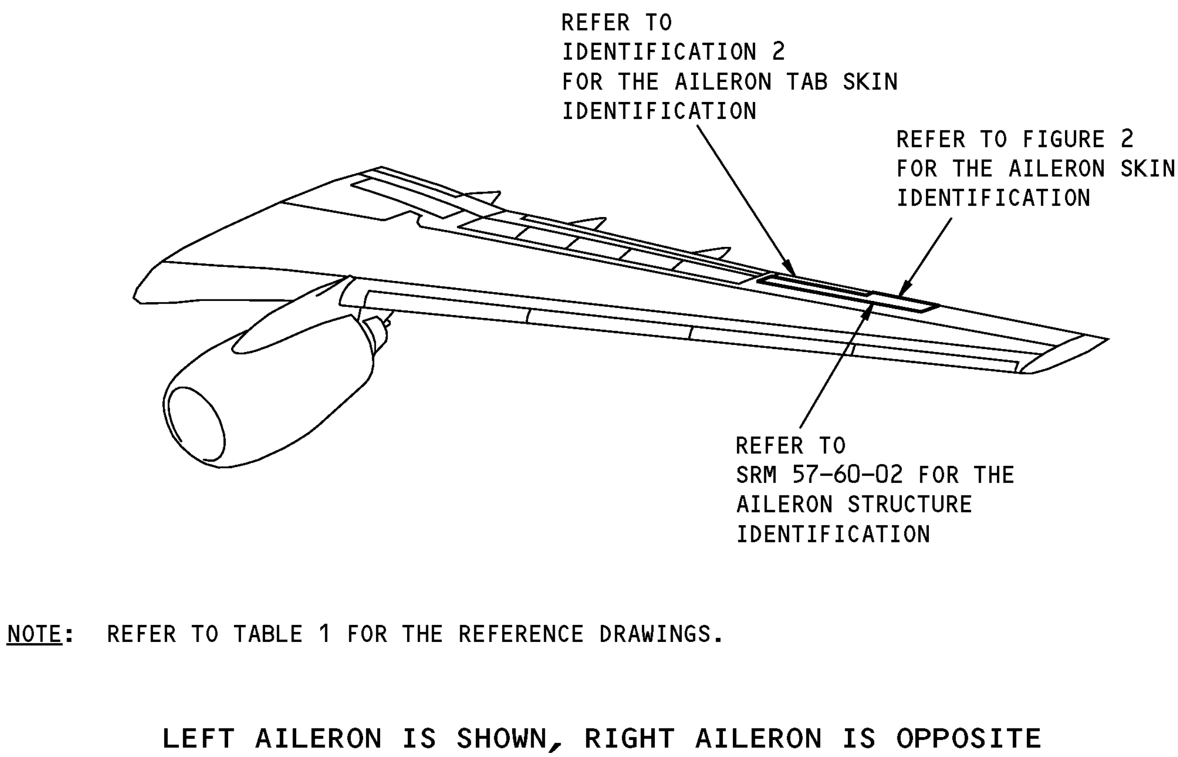
- REFER TO 57-51-00 FOR TRAILING EDGE.
- REFER TO 57-53-00 FOR TRAILING EDGE FLAPS.
- REFER TO 57-70-00 FOR SPOILERS.

**Aileron Diagram**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 110A1502 | Structures Diagram, Wing - Model 737-X |
| 113A7001 | Aileron Installation |
| 113A7002 | Aileron/Tab Assembly |

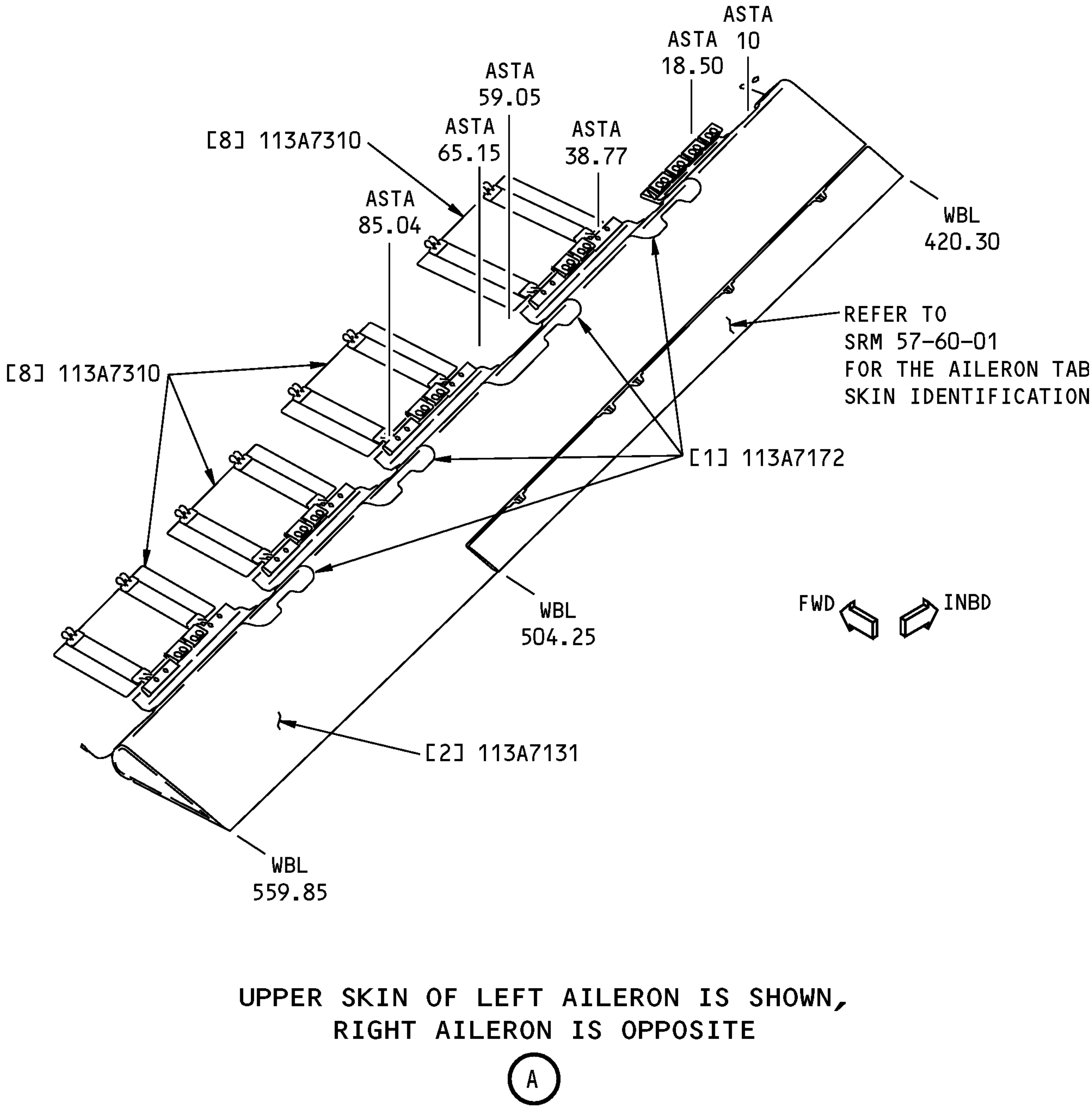
## IDENTIFICATION 1 - AILERON SKIN




**Aileron Skin Identification**
**Figure 1**

**Table 1**:

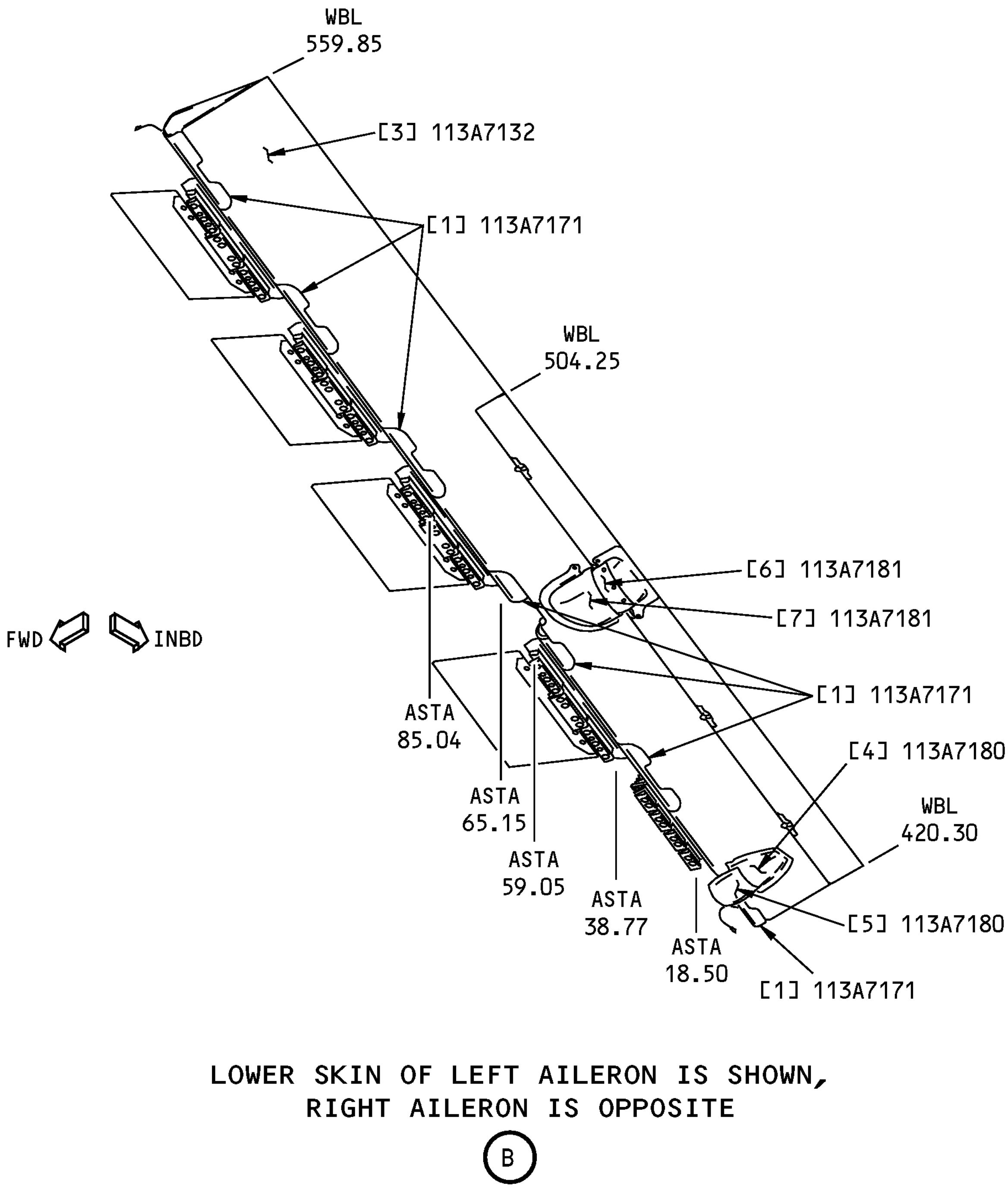
| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A7001 | Aileron Installation |
| 113A7002 | Aileron/Tab Assembly |
| 113A7100 | Aileron Assembly |
| 113A7200 | Tab Assembly - Aileron |
| 113A7300 | Balance Panel Assembly - Aileron |
| 113A7315 | Panel Assembly - Balance Bay, Aileron |




NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Aileron Skin Identification**
**Figure 2 (Sheet 1 of 2)**

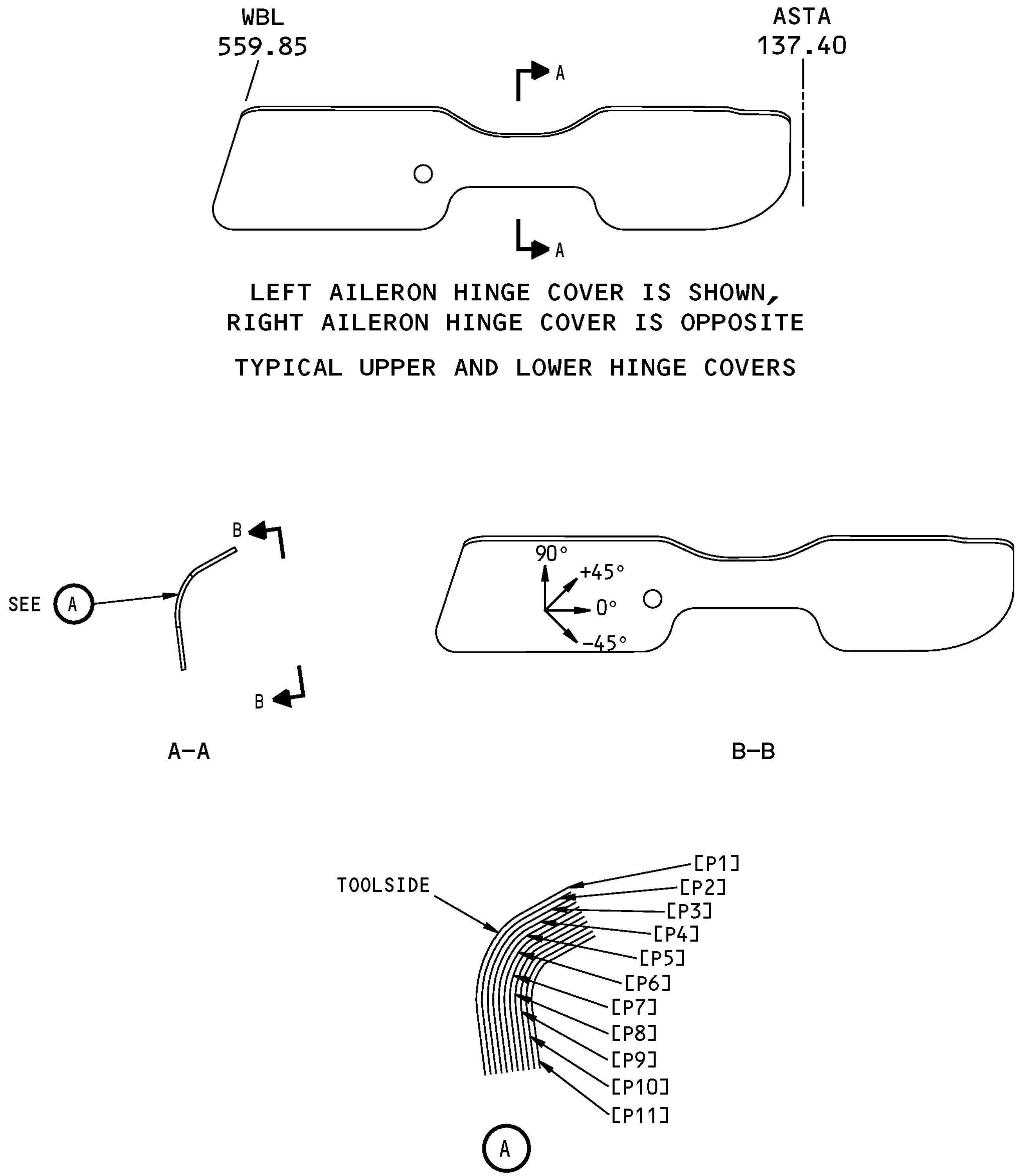



**Aileron Skin Identification**
**Figure 2 (Sheet 2 of 2)**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Bonded Hinge Cover, Upper and Lower | | Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: GFRP as given in BMS 8-139, Class I, Style 7781) Refer to the production drawing for the different hinge cover ply layups. Refer to Figure 3 for a typical hinge cover ply layup | |
| [2] | Bonded Upper Skin Panel | | Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW | |
| | Skin | | Refer to Figure 4 | |
| | Core (Between the outboard closure rib and the tab cutout rib) | 0.400 (1.02) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| | Core (Between the tab cutout rib and ASTA 65.05) | 0.400 (1.02) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| | Core (ASTA 59.15 to ASTA 11.31) | 0.400 (1.02) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [3] | Bonded Lower Skin Panel | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW | |
| | Skin | | Refer to Figure 5 | |
| | Core (Between the outboard closure rib and the tab cutout rib) | 0.400 (1.02) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| | Core (Between the tab cutout rib and ASTA 65.05) | 0.400 (1.02) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| | Core (ASTA 59.15 to ASTA 11.31) | 0.400 (1.02) | Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 | |
| [4] | Fixed Fairing - Actuator, Aileron | | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 6 | |
| [5] | Removable Fairing- Actuator, Aileron | | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 7 | |
| [6] | Removable Fairing - Control Rod, Aileron | | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 8 | |
| [7] | Fixed Fairing - Control Rod, Aileron | | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 9 | |
| [8] | Balance Panel (4) | 0.071 (1.80) | 7075-T73 sheet as given in QQ-A-250/12 | |

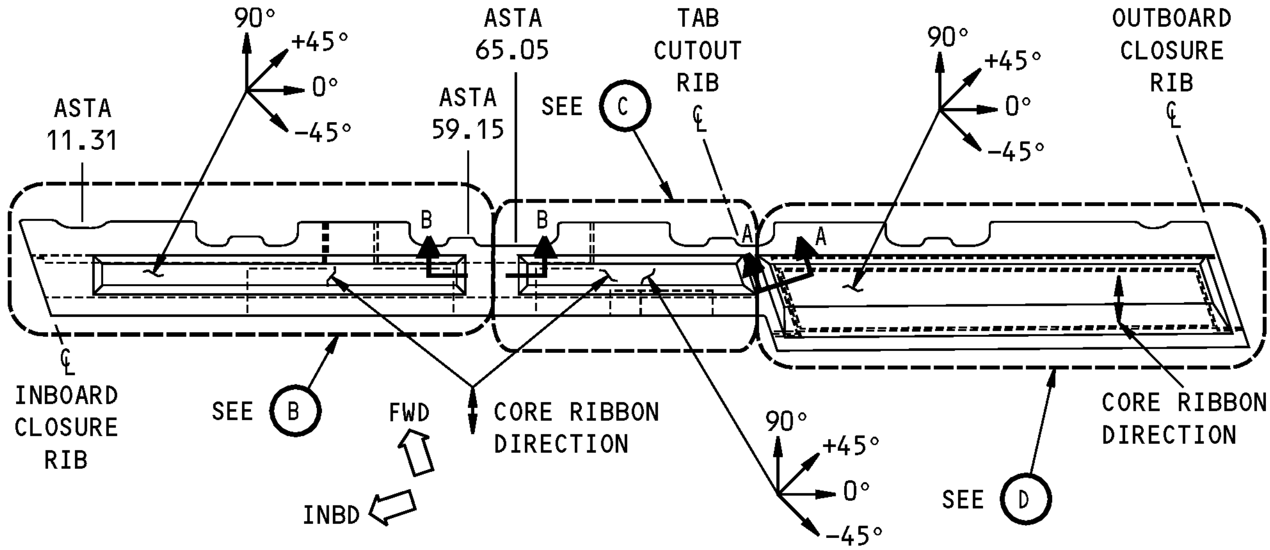
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




**Ply Direction, Core Ribbon Direction, and Ply Sequence for Figure 2, Item [1]**
**Figure 3**

**Table 3**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 3 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P5, P7, P9, P11 | + or - 45 degrees | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: GFRP as given in BMS 8-139, Class I, Style 7781) |
| P2, P4, P6, P8, P10 | 0 or 90 degrees | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: GFRP as given in BMS 8-139, Class I, Style 7781) |

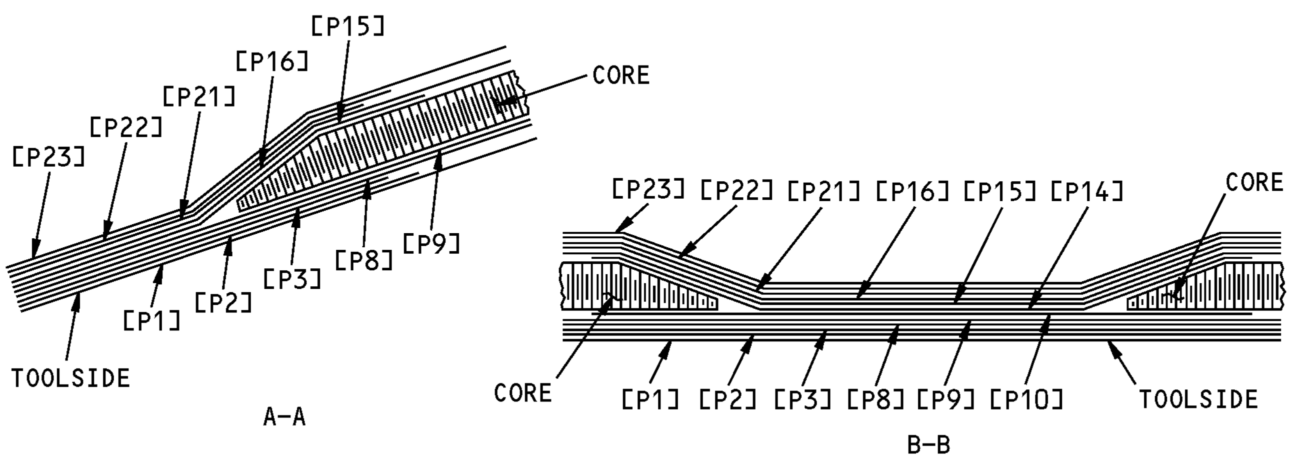



LEFT AILERON SKIN IS SHOWN, RIGHT AILERON SKIN IS OPPOSITE

VIEW IS ON THE BAGSIDE (NON-AERODYNAMIC) SURFACE

PLY LAYUP AND CORE RIBBON DIRECTION

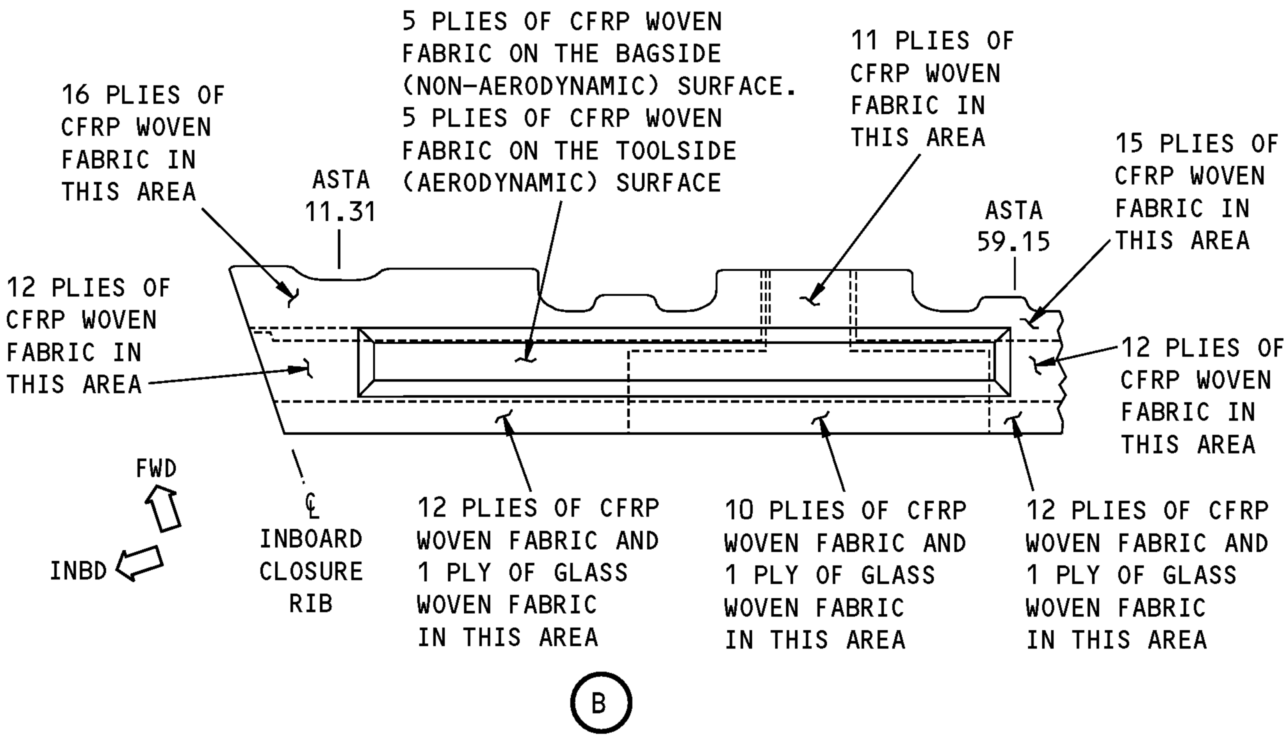
A

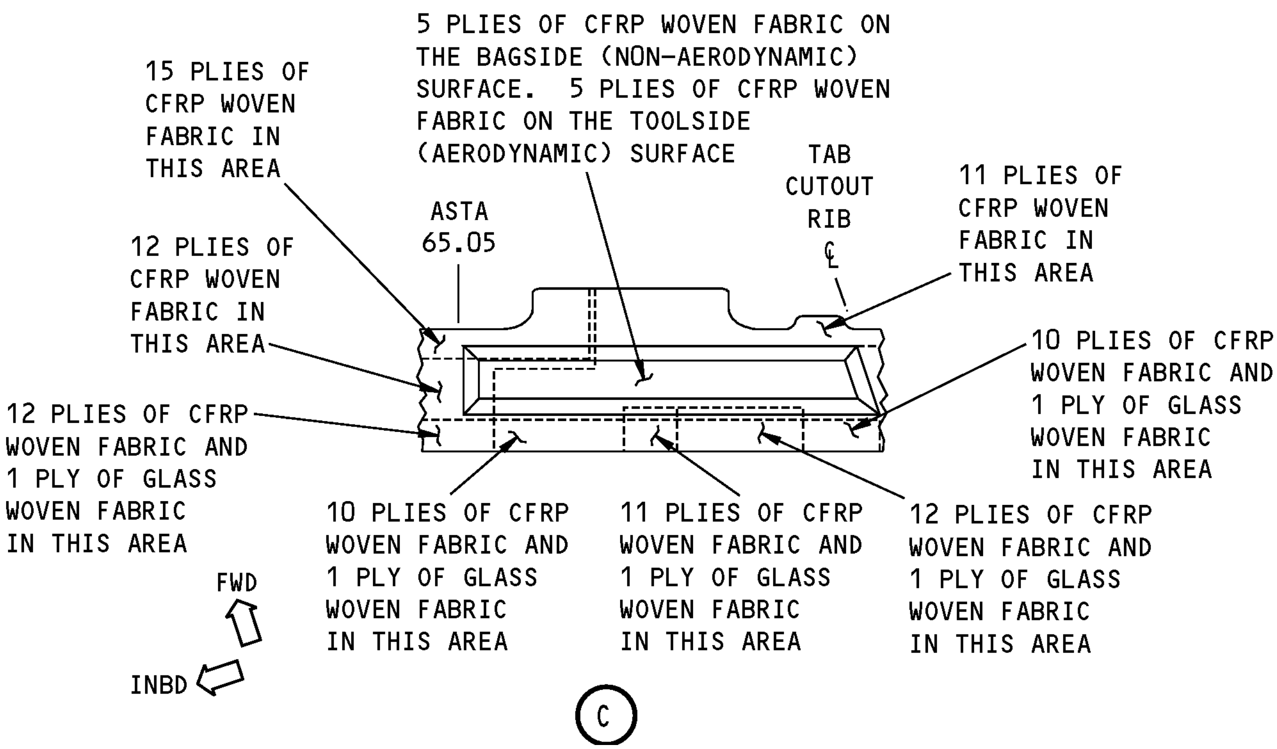



NOTES

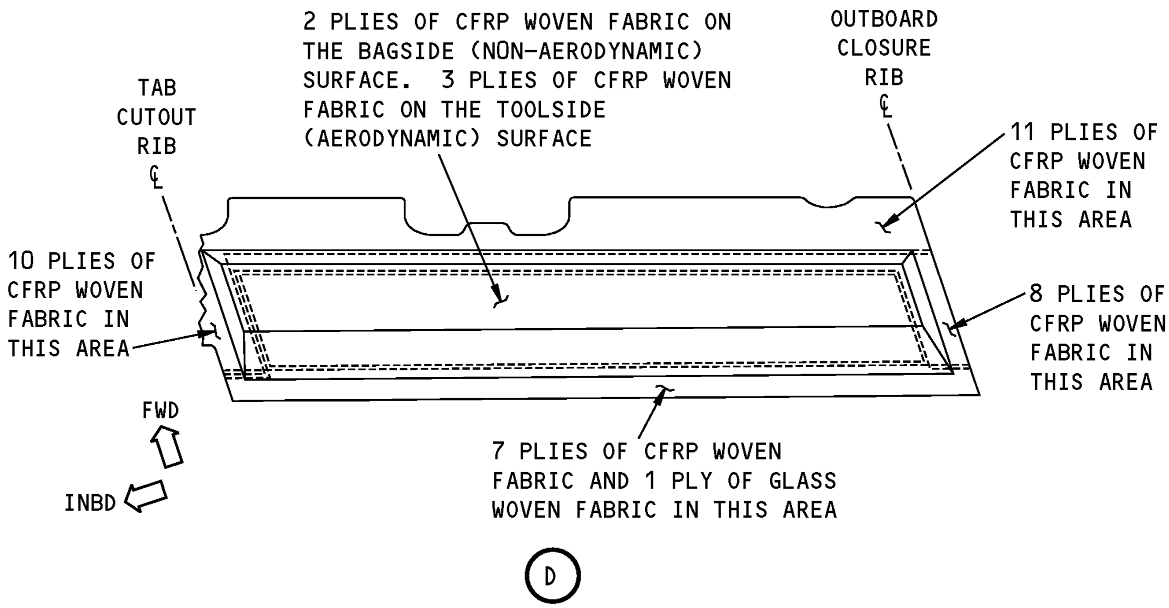
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS. REFER TO THE ENGINEERING DRAWING FOR MORE DATA.
- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN A-A AND B-B.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Upper Skin Panel, Figure 2, Item [2]**
**Figure 4 (Sheet 1 of 3)**



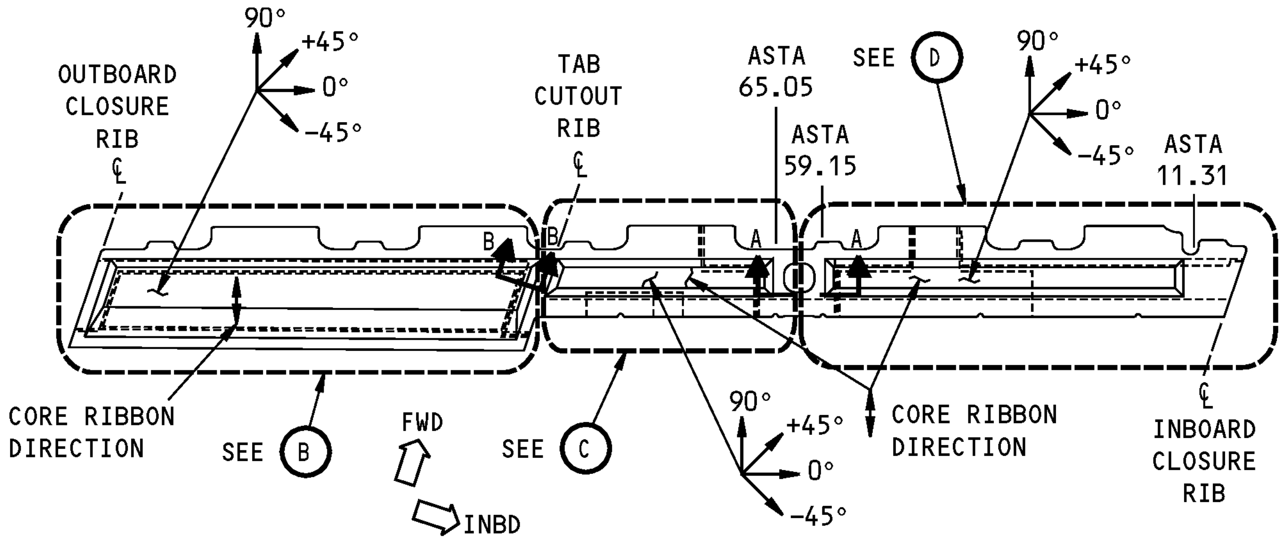

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Upper Skin Panel, Figure 2, Item [2]**
**Figure 4 (Sheet 2 of 3)**




**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Upper Skin Panel, Figure 2, Item [2]**
**Figure 4 (Sheet 3 of 3)**

**Table 4**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 4 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P7, P9, P15, P17, P21, P23 | + or - 45 Degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P4, P5, P6, P8, P10, P11, P12, P13, P14, P16, P18, P19, P20, P22 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P24 | Optional | GFRP woven fabric as given in BMS 8-139, Class III, Style 108 |

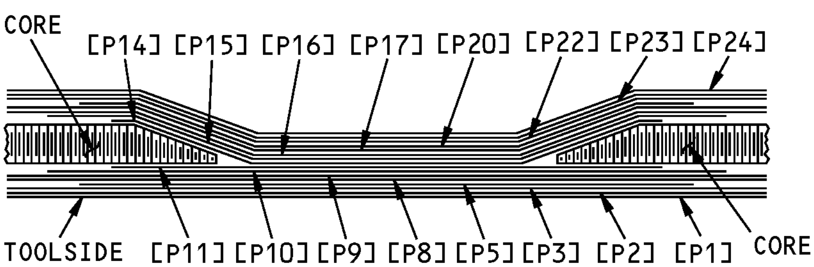



LEFT AILERON SKIN IS SHOWN, RIGHT AILERON SKIN IS OPPOSITE

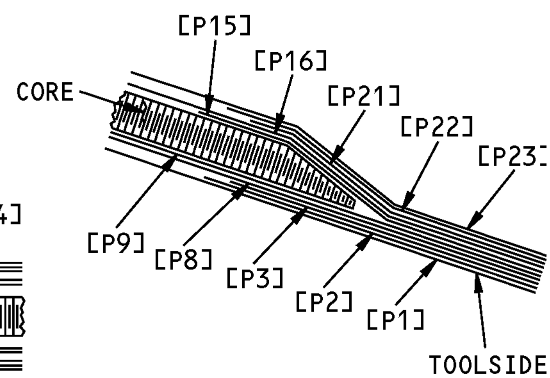
VIEW IS ON THE BAGSIDE (NON-AERODYNAMIC) SURFACE

PLY LAYUP AND CORE RIBBON DIRECTION

A




A-A

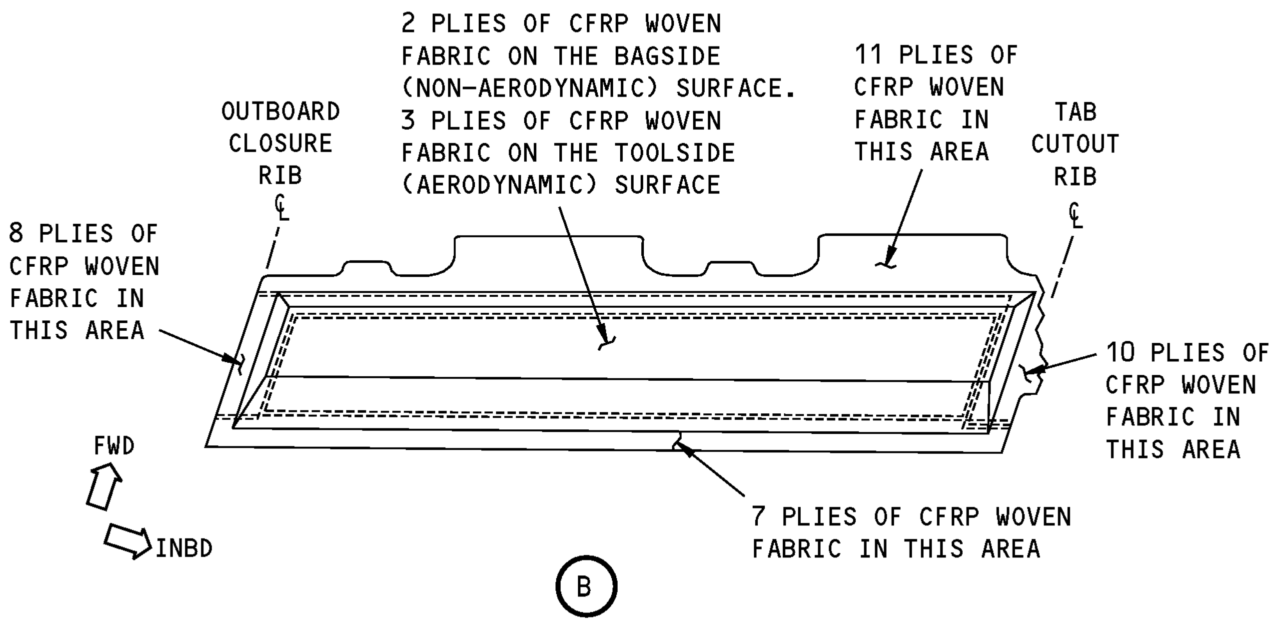


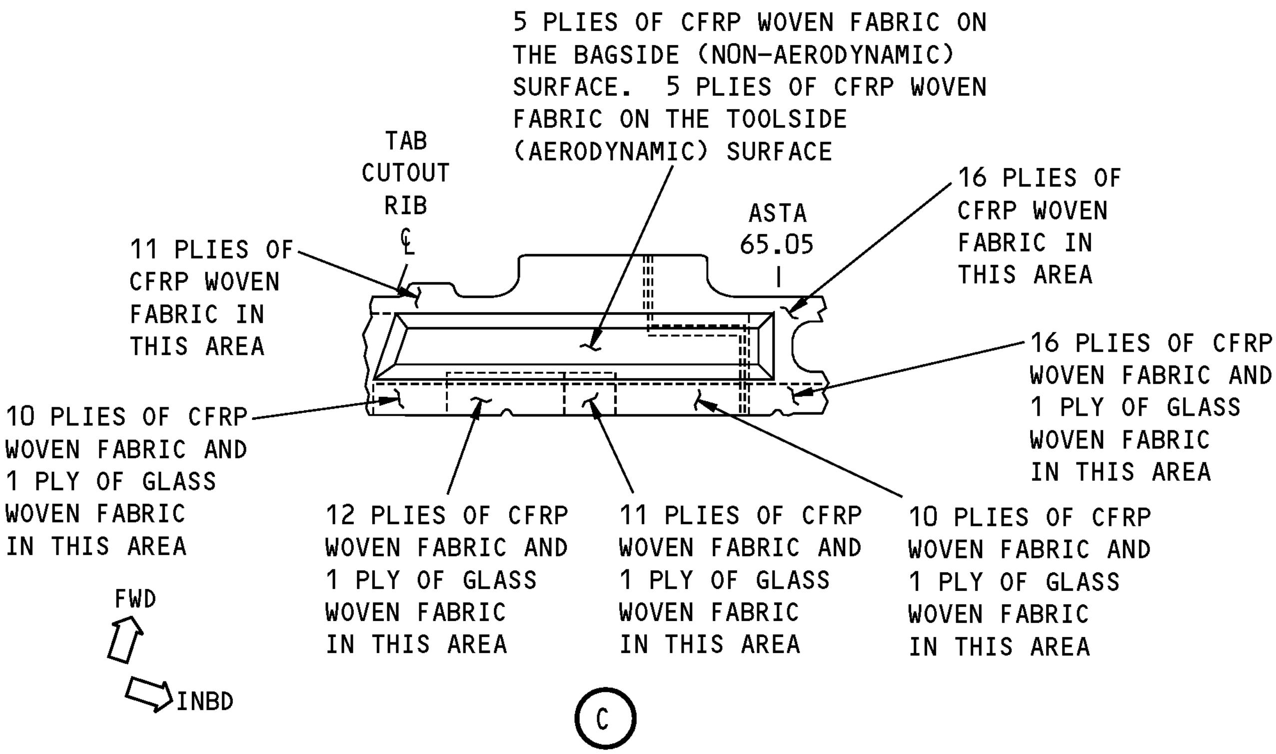

B-B

NOTES

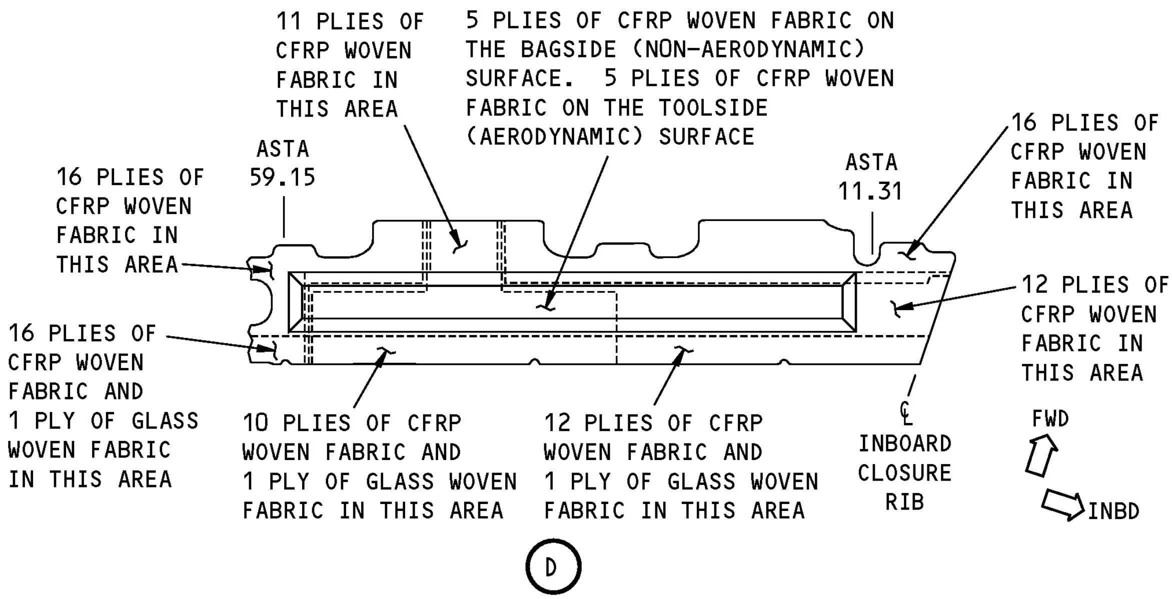
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS. REFER TO THE ENGINEERING DRAWING FOR MORE DATA.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN A-A AND B-B.

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Lower Skin Panel, Figure 2, Item [3]**
**Figure 5 (Sheet 1 of 3)**



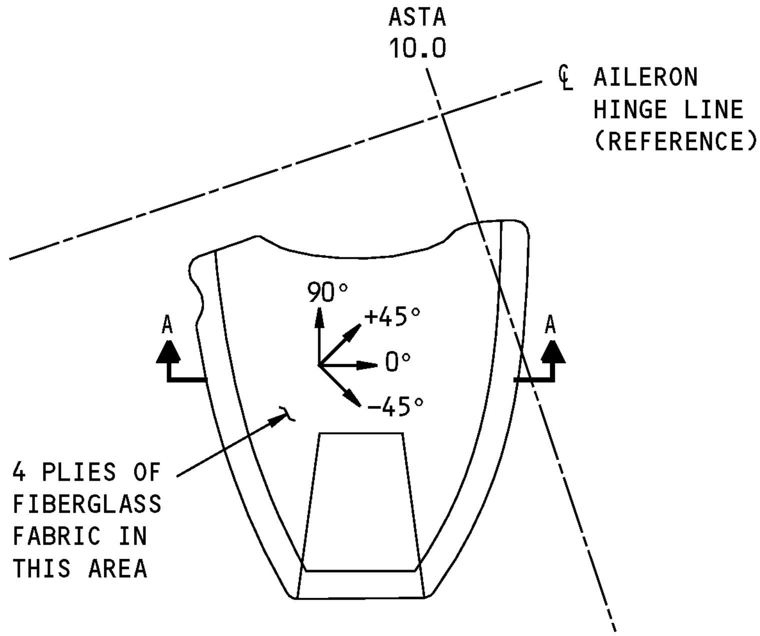

**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Lower Skin Panel, Figure 2, Item [3]**
**Figure 5 (Sheet 2 of 3)**




**Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Lower Skin Panel, Figure 2, Item [3]**
**Figure 5 (Sheet 3 of 3)**

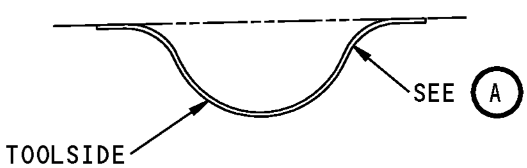
**Table 5**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 5 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P5, P7, P9, P10, P15, P16, P18, P20, P22, P24 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P4, P6, P8, P11, P12, P13, P14, P17, P19, P21, P23 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P25 | Optional | GFRP woven fabric as given in BMS 8-139, Class III, Style 108 |

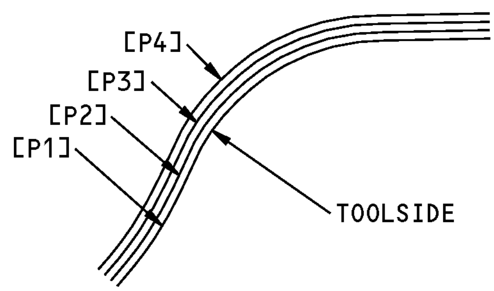



LEFT AILERON FAIRING SHOWN, RIGHT AILERON FAIRING OPPOSITE

FIXED FAIRING - AILERON ACTUATOR




A-A




A

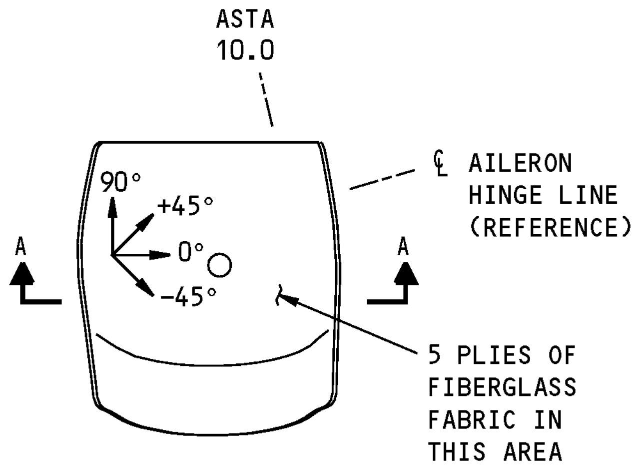
NOTES

- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

**Ply Direction and Ply Sequence for Figure 2, Item [4]**
**Figure 6**

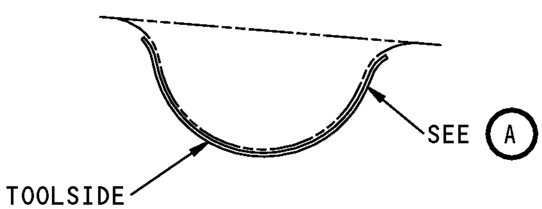
**Table 6**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 6 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 through P4 | 0 or 90 degrees | GFRP woven fabric as given in BMS 8-139, Class III, Grade B, Style 1581 |

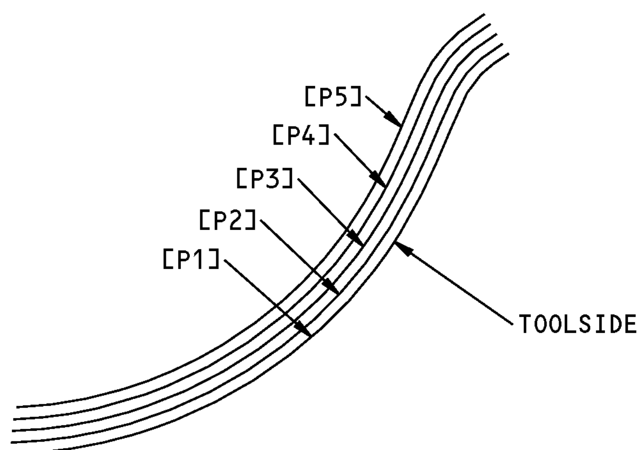



LEFT AILERON FAIRING SHOWN, RIGHT AILERON FAIRING OPPOSITE

FIXED FAIRING - AILERON ACTUATOR




A-A




A

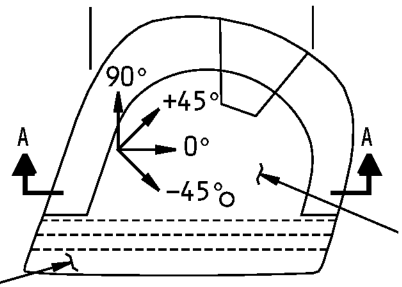
NOTES

- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

**Ply Direction and Ply Sequence for Figure 2, Item [5]**
**Figure 7**

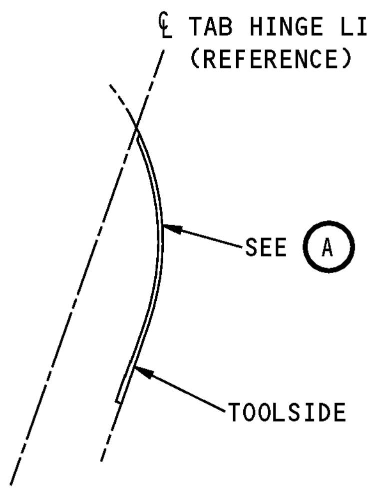
**Table 7**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 7 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 through P5 | 0 or 90 degrees | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 |

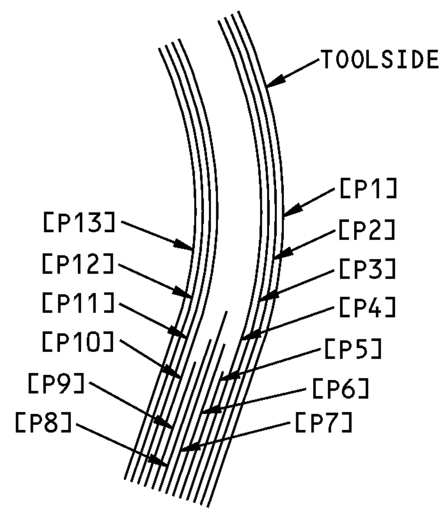



LEFT AILERON FAIRING SHOWN, RIGHT AILERON FAIRING OPPOSITE

REMOVABLE FAIRING - AILERON CONTROL ROD




A-A




A

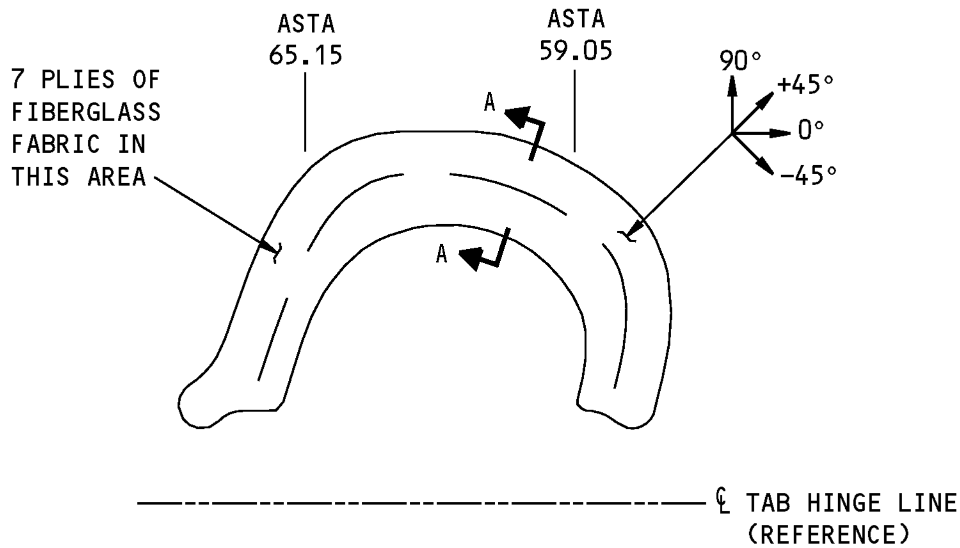
NOTES

- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

**Ply Direction and Ply Sequence for Figure 2, Item [6]**
**Figure 8**

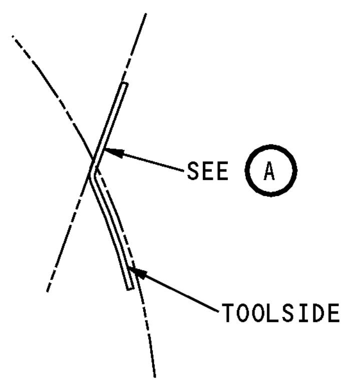
**Table 8**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 8 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P4, P7, P10, P13 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P3, P5, P6, P8, P9, P11, P12 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

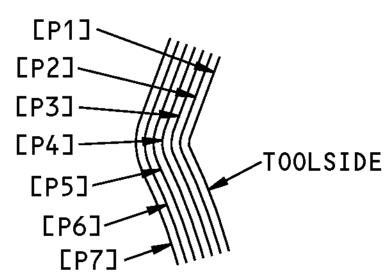



LEFT AILERON FAIRING SHOWN, RIGHT AILERON FAIRING OPPOSITE

FIXED FAIRING - AILERON CONTROL ROD




A-A




A

NOTES

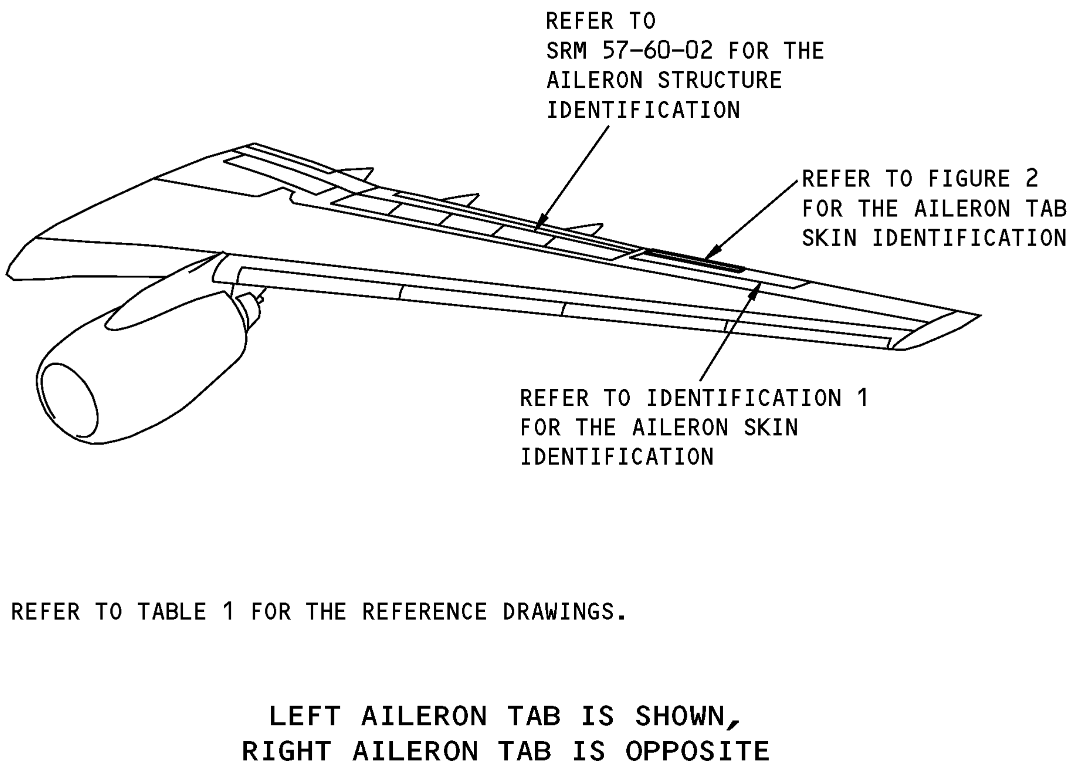
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

**Ply Direction and Ply Sequence for Figure 2, Item [7]**
**Figure 9**

**Table 9**:

| Y MATERIAL AND DIRECTION FOR FIGURE 9 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 through P7 | 0 or 90 degrees | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 |

## IDENTIFICATION 2 - AILERON TAB SKIN

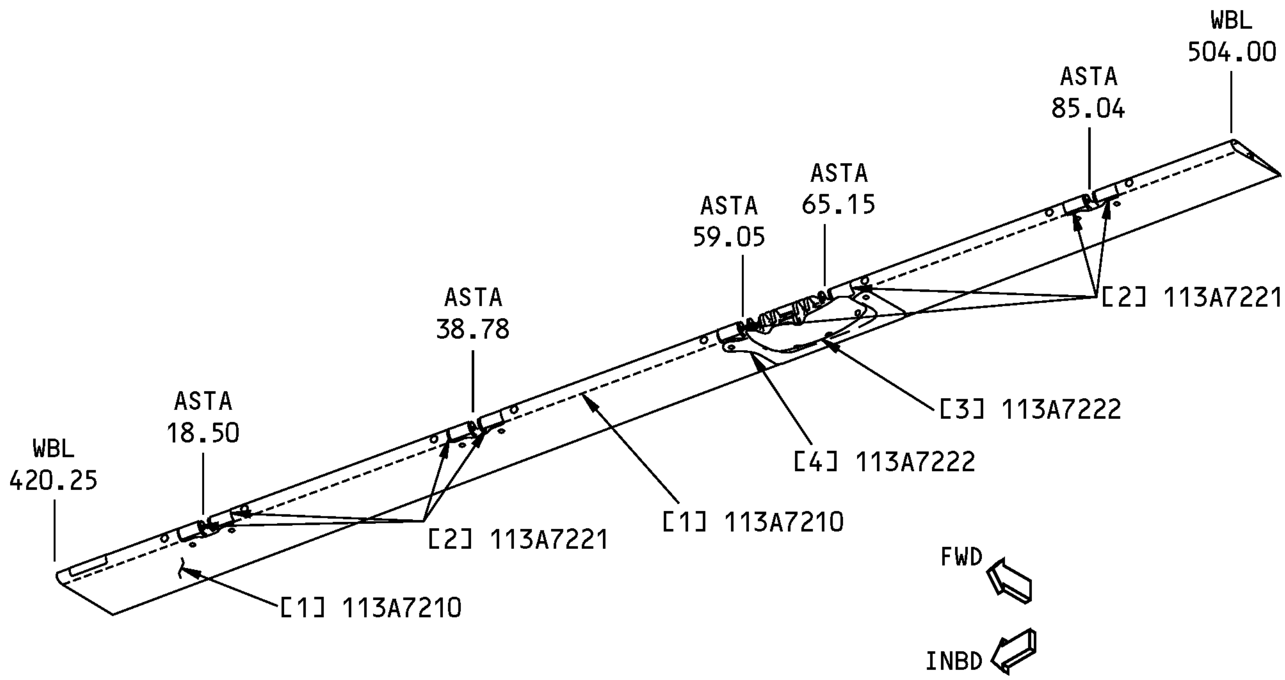



**Aileron/Tab Skin Identification**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 113A7001 | Aileron Installation |
| 113A7002 | Aileron/Tab Assembly |
| 113A7200 | Tab Assembly - Aileron |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A7300 | Balance Panel Assembly - Aileron |
| 113A7315 | Panel Assembly - Balance Bay, Aileron |



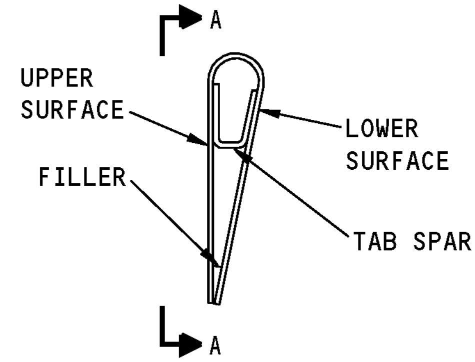

NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Aileron Tab Skin Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Bonded Tab Part | | Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW | |
| | Skin | | Refer to Figure 3 | |
| | Tab Spar | | Refer to Figure 4 | |
| [2] | Hinge Cover | | Nylon injection molding resin as given in BMS 8-323, Type I, Class 3, Grade 30, Form B. Color same as BAC 707 Gray | |
| [3] | Removable Fairing - Aileron Tab Control Rod | | Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 5 | |
| [4] | Fixed Fairing - Aileron Tab Control Rod | | GFRP as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 6 | |

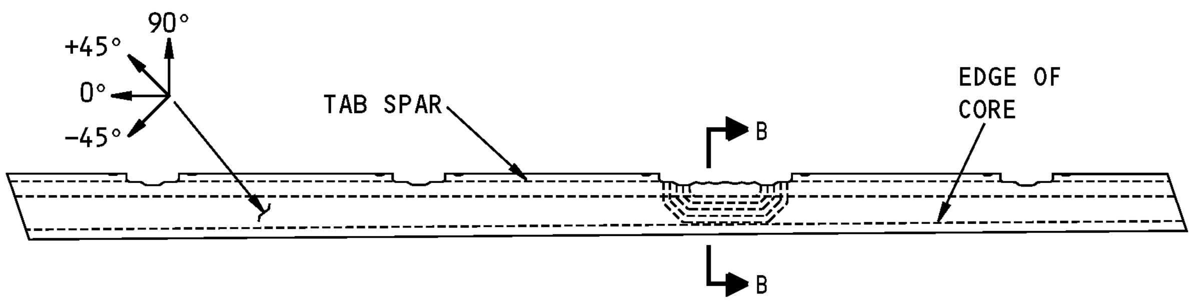
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



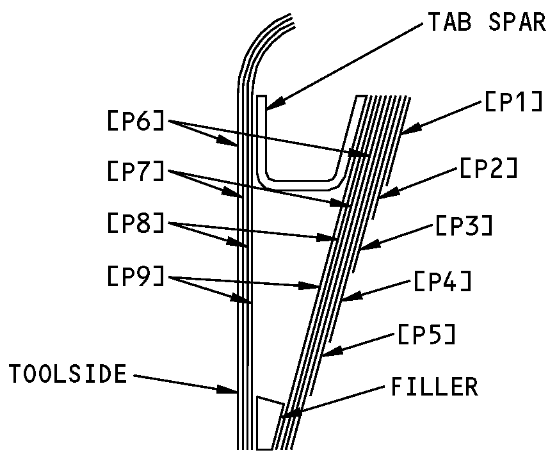

LEFT AILERON TAB IS SHOWN, RIGHT AILERON TAB IS OPPOSITE

OUTBOARD VIEW OF BONDED AILERON TAB SKIN

A




A-A




B-B

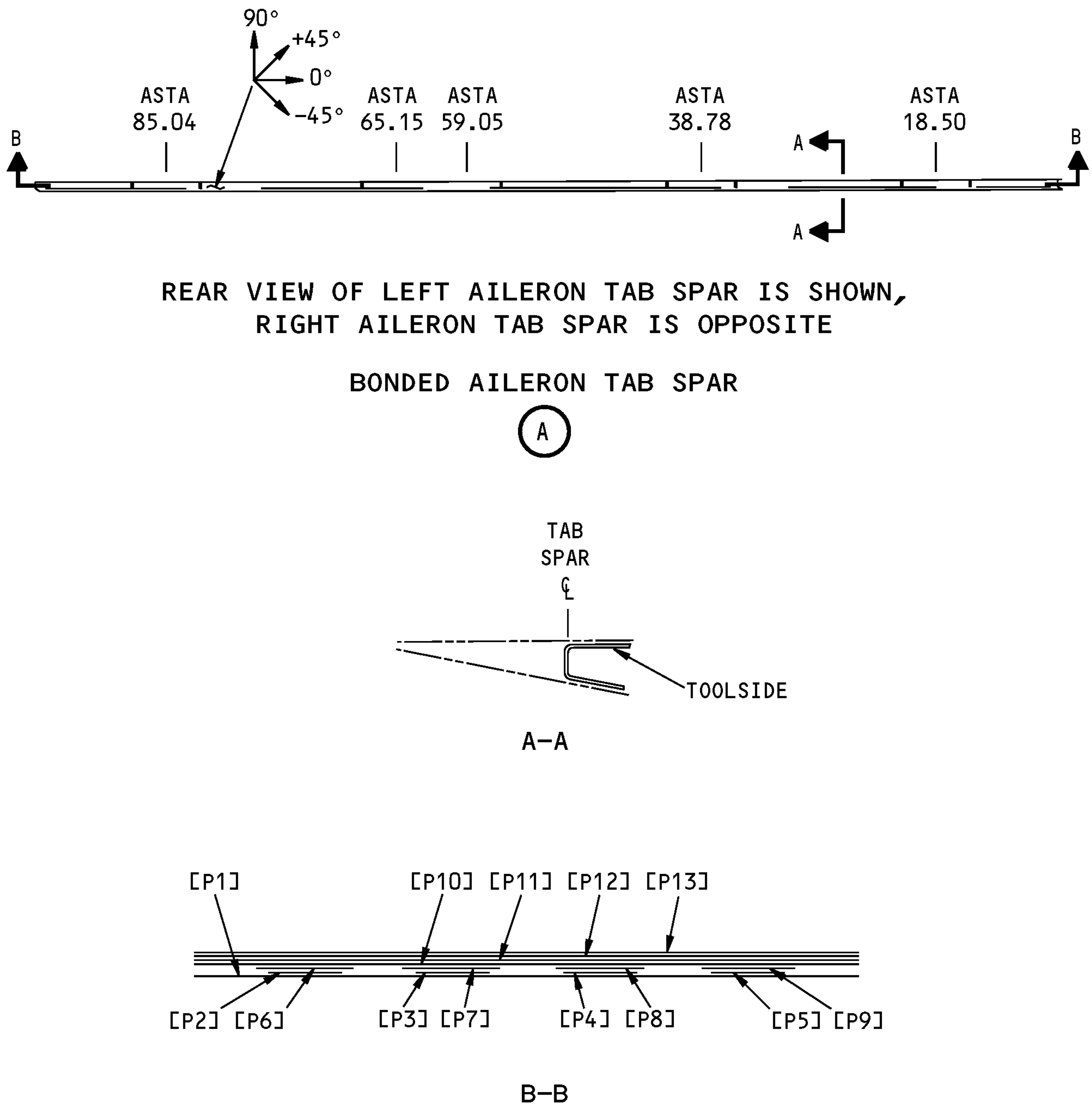
NOTES

- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN B-B.

**Ply Direction and Ply Sequence for the Skin, Figure 2, Item [1]**
**Figure 3**

**Table 3**:

| PLY MATERIAL AND DIRECTION FOR THE AILERON TAB SKIN, FIGURE 2, ITEM [1] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P4, P6, P9 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P3, P5, P7, P8 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |



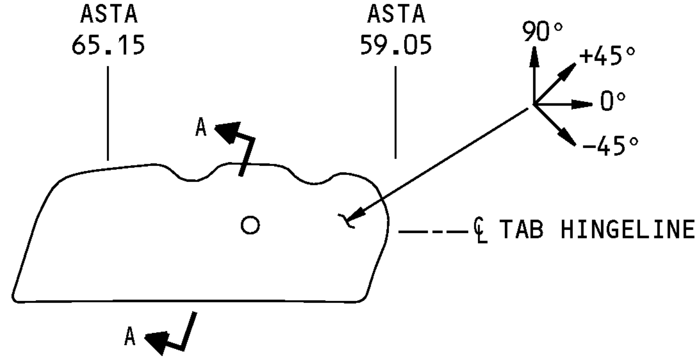

NOTES

- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN B-B.

**Ply Direction and Ply Sequence for the Tab Spar, Figure 2, Item [1]**
**Figure 4**

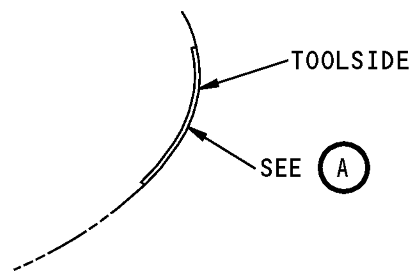
**Table 4**:

| PLY MATERIAL AND DIRECTION FOR THE AILERON TAB SPAR, FIGURE 2, ITEM [1] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | Optional | GFRP woven fabric as given in BMS 8-79, Class III, Style 108 |
| P2 through P10, P13 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P11, P12 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

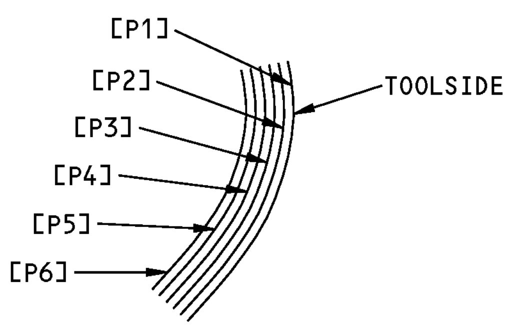



LEFT TAB FAIRING IS SHOWN, RIGHT TAB FAIRING IS OPPOSITE

REMOVABLE FAIRING - AILERON TAB CONTROL ROD




A-A




A

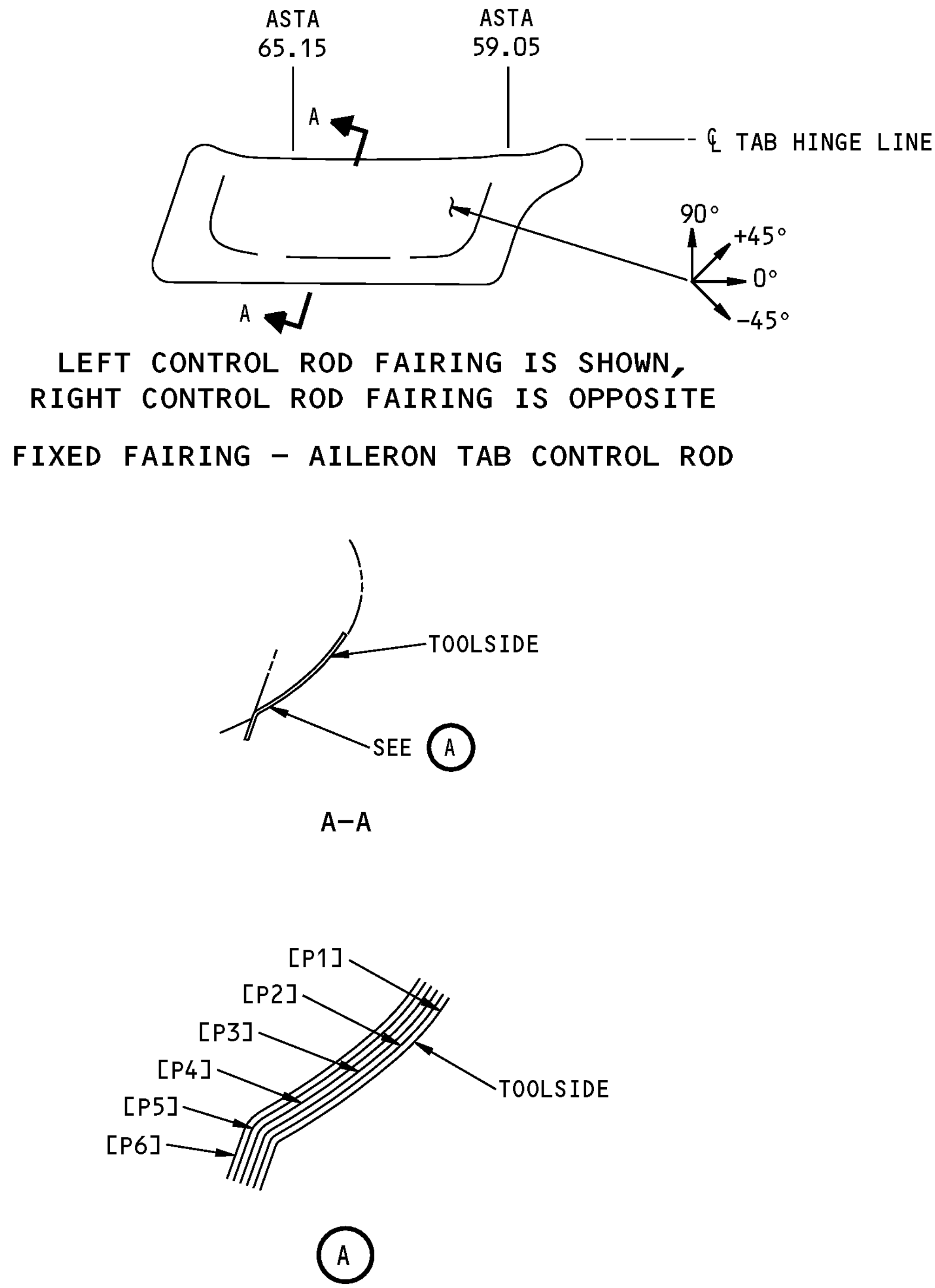
NOTES

- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

**Ply Direction and Ply Sequence for the Removable Fairing, Figure 2, Item [3]**
**Figure 5**

**Table 5**:

| PLY MATERIAL AND DIRECTION FOR THE REMOVABLE FAIRING, FIGURE 2, ITEM [3] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 through P6 | 0 or 90 degrees | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 |




NOTES

- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

**Ply Direction and Ply Sequence for Fixed Fairing, Figure 2, Item [4]**
**Figure 6**

**Table 6**:

| PLY MATERIAL AND DIRECTION FOR THE FIXED FAIRING, FIGURE 2, ITEM [4] | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 through P6 | 0 or 90 degrees | GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 |

## ALLOWABLE DAMAGE 1 - AILERON SKIN

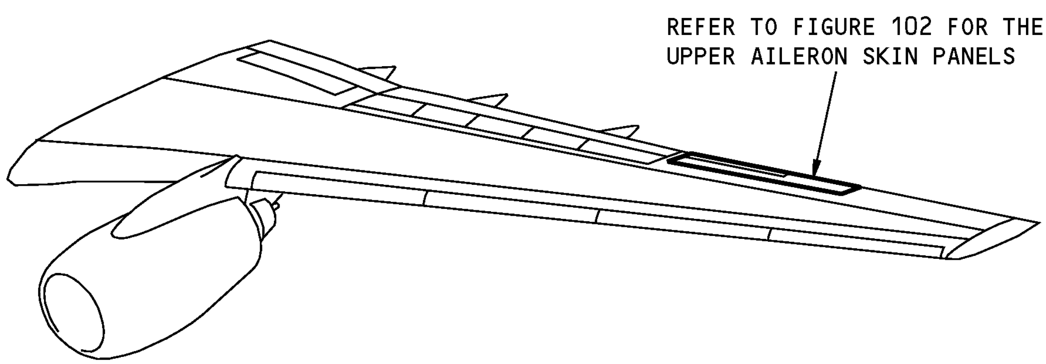
### 1. Applicability

A. This subject gives the allowable damage limits for the upper and lower aileron skins shown in Wing Aileron Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.

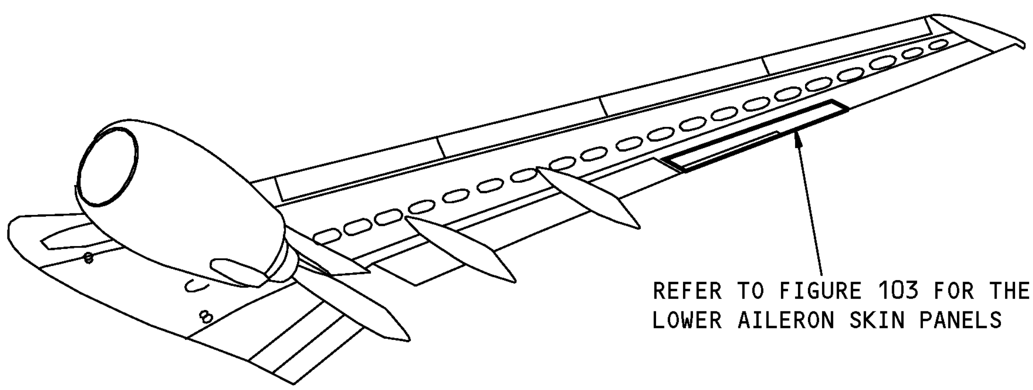
B. For the allowable damage zones of the aileron, refer to:

(1) Aileron Upper Skin Zones, Figure 102/ALLOWABLE DAMAGE 1 for the aileron upper skin

(2) Aileron Lower Skin Zones, Figure 103/ALLOWABLE DAMAGE 1 for the aileron lower skin.




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Aileron Skin Panel Location**
**Figure 101**

2. **General**

A. Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) For the honeycomb core areas, the tap test is an alternative procedure to an instrumented NDT.

(2) Refer to Damage Definitions, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(3) Refer to Definitions of the Facesheets, Figure 105/ALLOWABLE DAMAGE 1 for the definitions of the facesheets of a honeycomb area.

B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two procedures that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) Make sure the tape is in satisfactory condition after each 400 flight hours interval or more frequently.

(d) Repair the damage at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area. Refer to 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Use a vacuum and heat to remove moisture from the solid laminate and honeycomb cells. Refer to 51-70-04.

(2) Make a temporary seal with aluminum foil tape (speed tape).

(3) Keep a record of the location.

(4) Repair the damage at or before 400 flight hours from the time the seal was made.

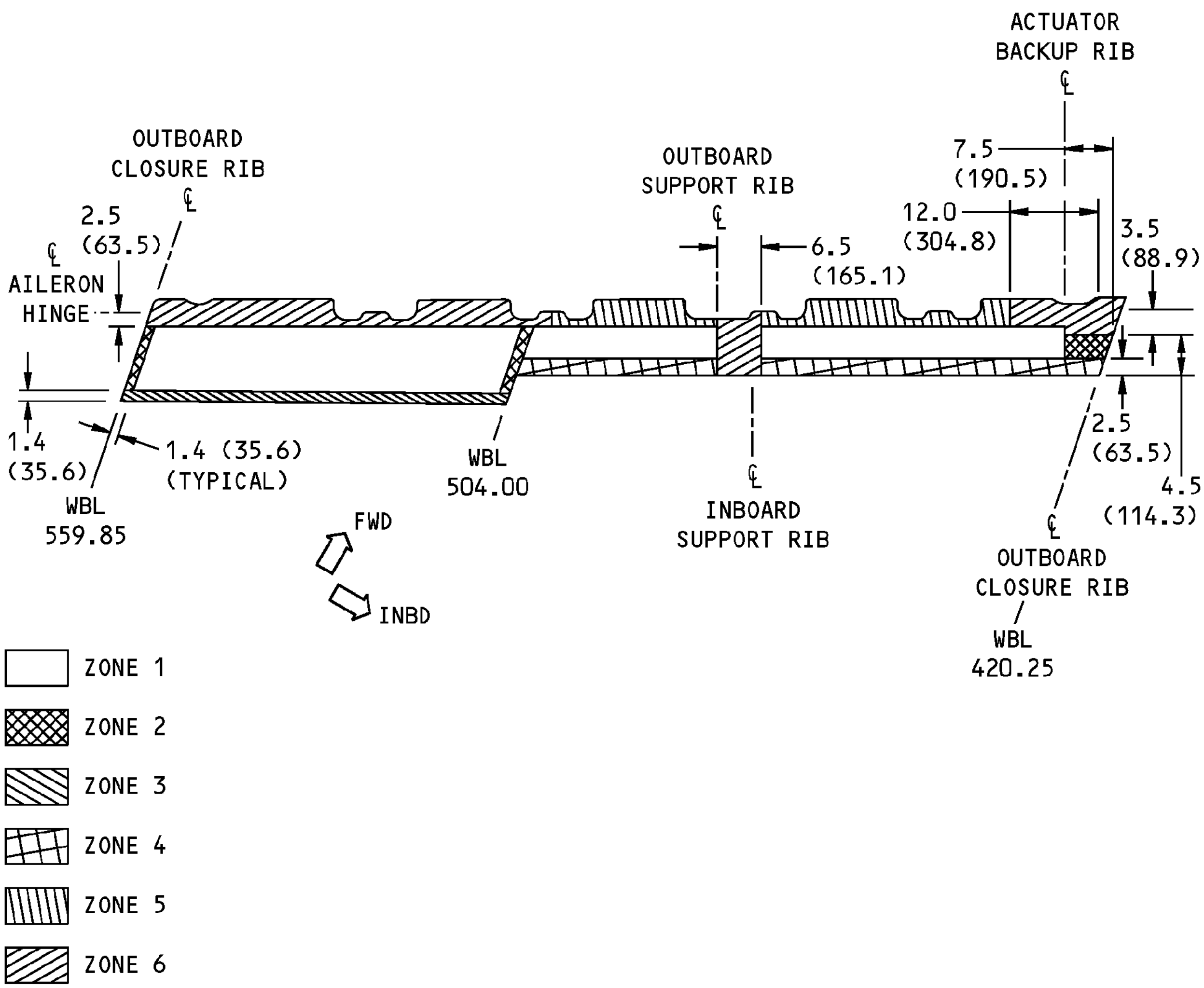
E. The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause carbon fiber damage and are sealed.

F. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| **TYPE OF STRUCTURE** | **ZONE LOCATION** | **PARAGRAPH** |
| HONEYCOMB CORE AREAS | 1 | 4.A |

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| **TYPE OF STRUCTURE** | **ZONE LOCATION** | **PARAGRAPH** |
| SOLID LAMINATE AREAS | 2 | 4.B |
| | 3 | 4.C |
| | 4 | 4.D |
| | 5 | 4.E |
| | 6 | 4.F |

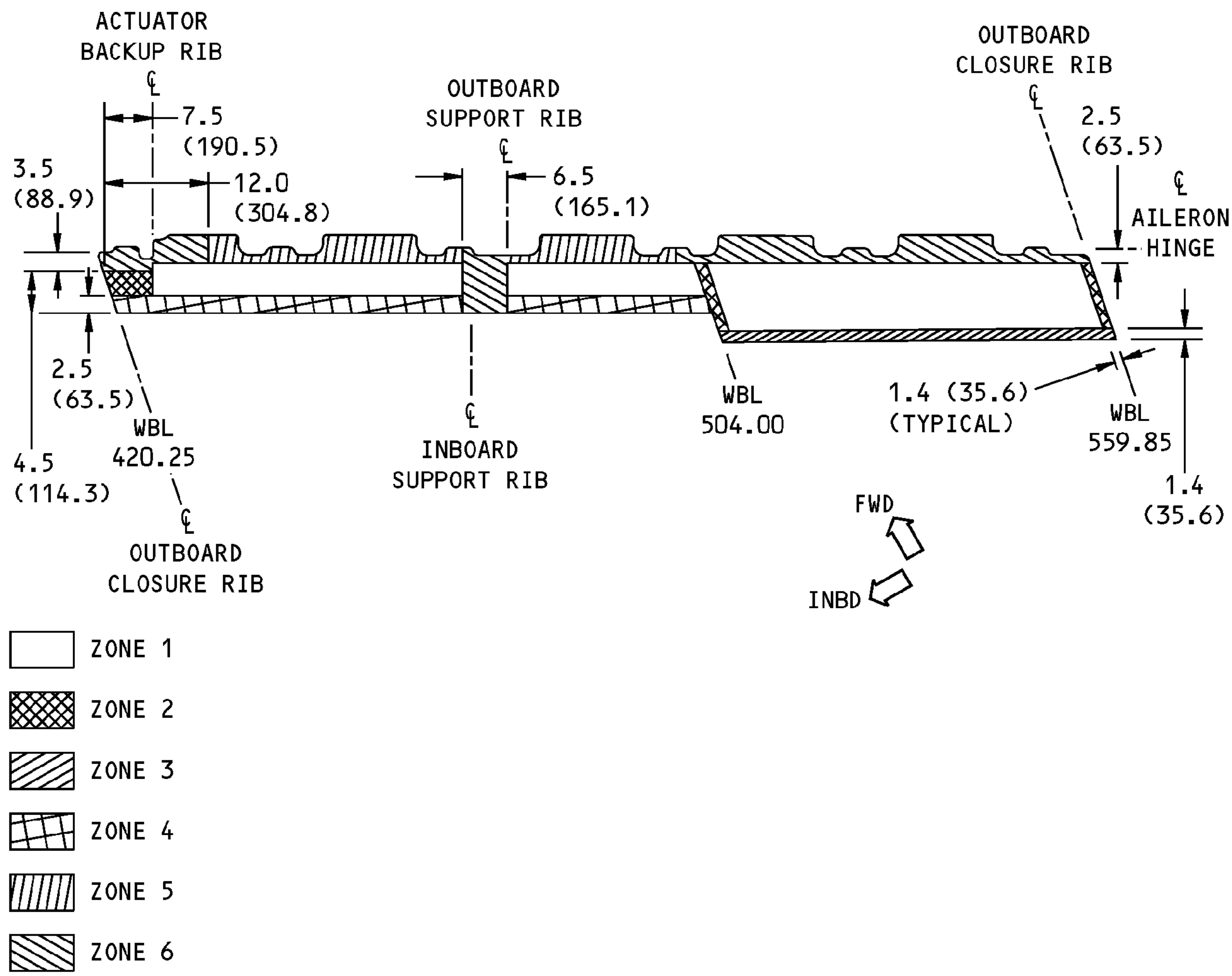



NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANELS

**Aileron Upper Skin Zones**
**Figure 102**

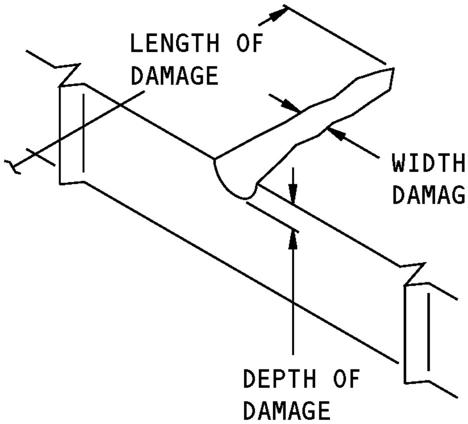



NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

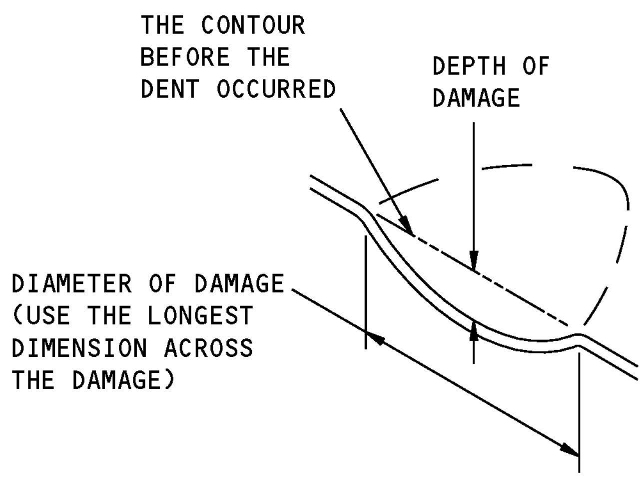
LOWER SKIN PANELS

**Aileron Lower Skin Zones**
**Figure 103**



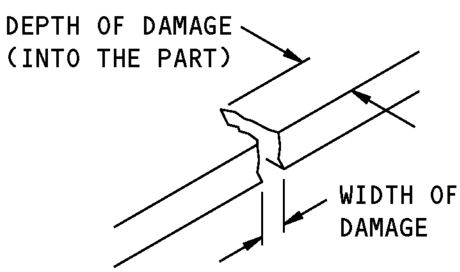

DEFINITIONS FOR
NICK, GOUGE, OR SCRATCH DAMAGE

A




DEFINITIONS FOR
DENT DAMAGE

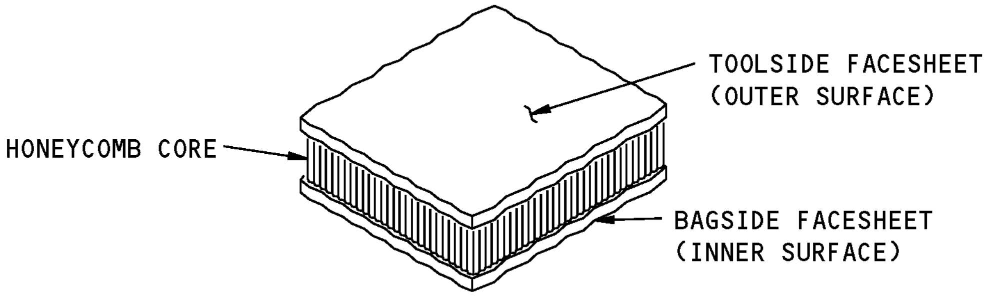
B




DEFINITIONS FOR
EDGE DAMAGE

C

**Damage Definitions**
**Figure 104**




**Definitions of the Facesheets**
**Figure 105**

## 3. References

| Reference | Title |
|---|---|
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-60-01 | AILERON BALANCING PROCEDURES |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

## 4. Allowable Damage Limits

A. Zone 1 - Honeycomb Core Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of one ply

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 2.0 inches (50.8 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a fitting.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage.

(4) Holes and Punctures are permitted if they are:
    (a) A maximum of 0.50 inch (12.7 mm) of one toolside facesheet and the core in depth
    (b) A maximum of 0.50 inch (12.7 mm) in diameter
    (c) A minimum of 3.0 inches (76.2 mm) away from other damage
    (d) A minimum of 3.0 inches (76.2 mm) away from a fitting.
(5) Delamination is permitted as shown in Definitions of the Facesheets, Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:
    (a) A maximum of 0.50 inch (12.7 mm) of one toolside facesheet in depth
    (b) A maximum of 0.50 inch (12.7 mm) in length
    (c) A maximum of 0.50 inch (12.7 mm) in width.

B. Zone 2 - Solid Laminate Areas
(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.
(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:
    (a) A maximum of 2.0 inches (50.8 mm) in length
    (b) A maximum of one ply

    **NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

    (c) A maximum of 0.25 inch (6.35 mm) in width
    (d) A minimum of 3.0 inches (76.2 mm) away from other damage.
(3) Dents:
    (a) Dents that do not cause damage to the carbon fibers are permitted if they are:
        1) A maximum of 0.50 inch (12.7 mm) in width
        2) A maximum of 0.05 inch (1.27 mm) in depth

        **NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

        3) A minimum of 3.0 inches (76.2 mm) away from other damage.
(4) Holes and Punctures are permitted if they are:
    (a) A maximum of 0.45 inch (11.43 mm) in diameter
    (b) A minimum of 3.0 inches (76.2 mm) away from other damage.
(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail A if it is:
    (a) A maximum of 2.0 inch (50.8 mm) in length
    (b) A maximum of 0.90 inch (22.86 mm) in width.
(6) Edge damage is permitted if it is:
    (a) A maximum of 0.06 inch (1.52 mm) in depth
    (b) A maximum of 0.25 inch (6.35 mm) in width
    (c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole
    (d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail A.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

C. Zone 3 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of 2.0 inches (50.8 mm) in length

(b) A maximum of one ply

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch 1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.60 inch (15.24 mm) in diameter

(b) A minimum of 3.0 inches (76.2 mm) away from other damage.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 1.50 inch (38.1 mm) in length

(b) A maximum of 0.90 inch (22.86 mm) in width.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.05 inch (1.27 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail B.

D. Zone 4 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of 2.0 inches (50.8 mm) in length

(b) A maximum of one ply

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage

(e) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage

4) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch (12.7 mm) in diameter

(b) A minimum of 3.0 inches (76.2 mm) away from other damage

(c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(5) Delamination is permitted as shown in Figure 107/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 0.50 inch (12.7 mm) in length

(b) A maximum of 0.50 inch (12.7 mm) in width

(c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.08 inch (2.03 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Figure 107/ALLOWABLE DAMAGE 1, Detail A

(e) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(7) Edge Erosion is permitted as shown in Figure 107/ALLOWABLE DAMAGE 1, Detail B.

E. Zone 5 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of 0.50 inch (12.7 mm) in length

(b) A maximum of one ply

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage

(e) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width
2) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage
4) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(4) Holes and Punctures are permitted if they are:
   (a) A maximum of 0.25 inch (6.35 mm) in diameter
   (b) A minimum of 3.0 inches (76.2 mm) away from other damage
   (c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

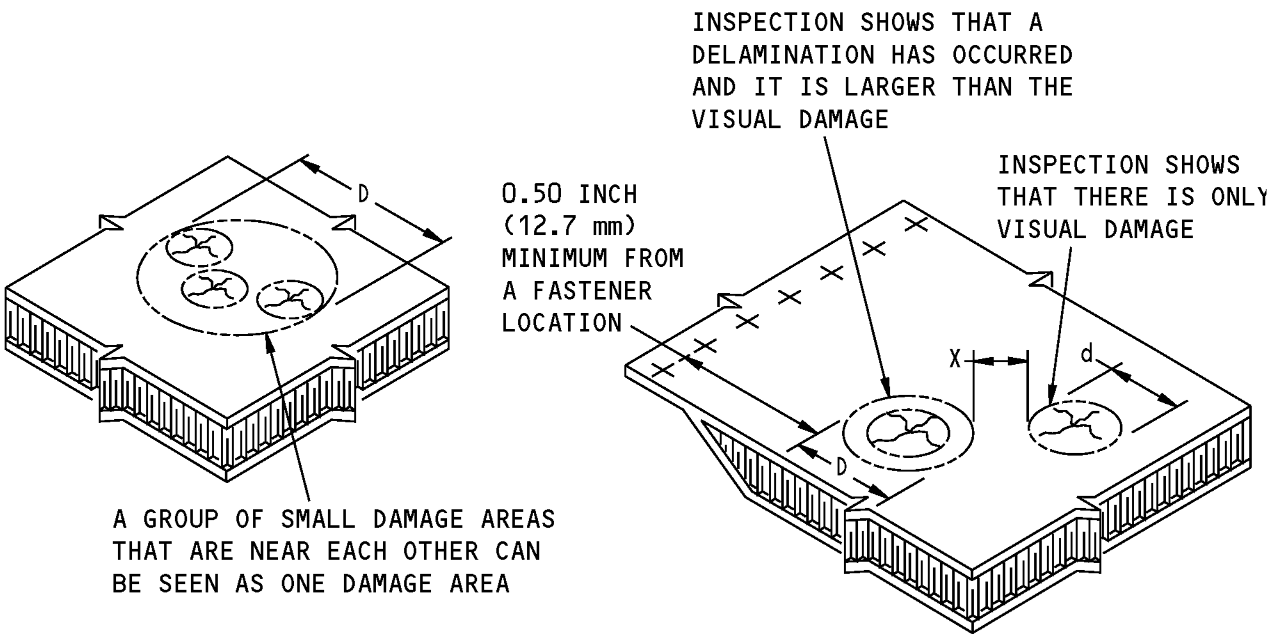
(5) Delamination is permitted as shown in Figure 107/ALLOWABLE DAMAGE 1, Detail A if it is:
   (a) A maximum of 0.50 inch (12.7 mm) in length
   (b) A maximum of 0.50 inch (12.7 mm) in width
   (c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(6) Edge damage is permitted if it is:
   (a) A maximum of 0.09 inch (2.29 mm) in depth
   (b) A maximum of 0.25 inch (6.35 mm) in width
   (c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole
   (d) A minimum distance from the edge of other damage as shown in Figure 107/ALLOWABLE DAMAGE 1, Detail B.
   (e) A minimum of 3.0 inches (76.2 mm) from a fitting.

(7) Edge Erosion is permitted as shown in Figure 107/ALLOWABLE DAMAGE 1, Detail B.

F. Zone 6 - Solid Laminate Areas

(1) Damage is not permitted.




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

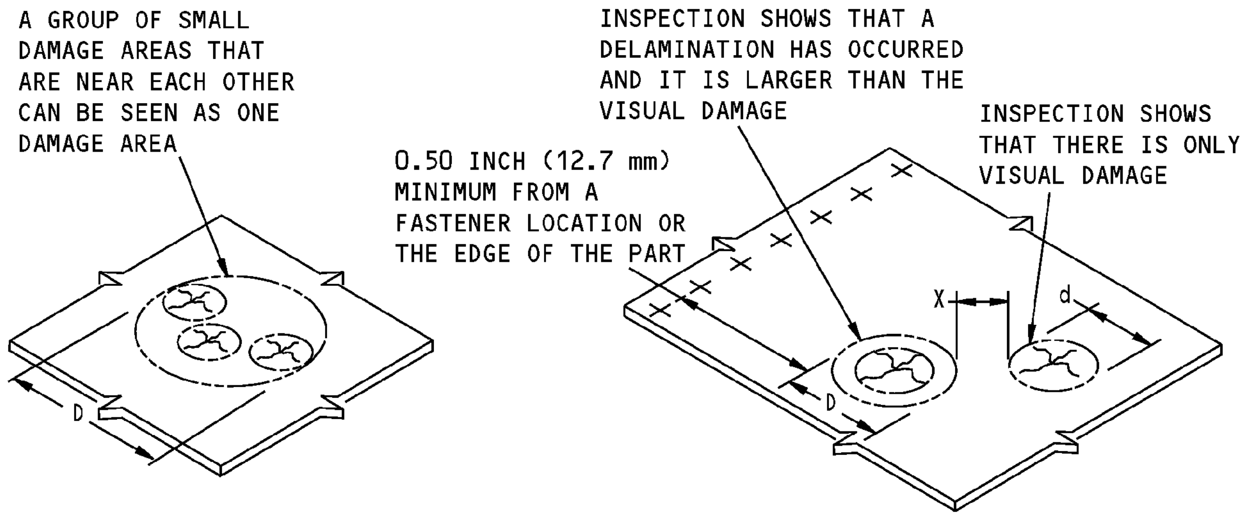
d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.0 INCHES (76.2 mm).

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

**Allowable Damage Limits**
**Figure 106**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

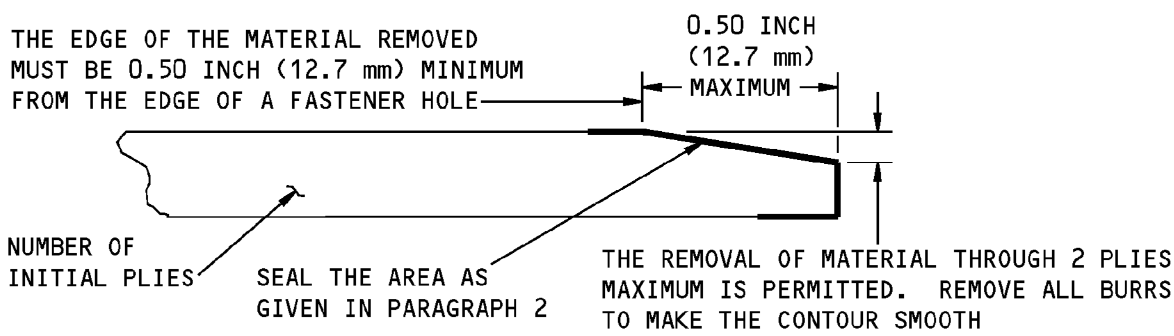
X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.0 INCHES (76.2 mm).

A MINIMUM OF 0.50 INCH (12.7 mm) FROM THE EDGE OF THE PART IS ALLOWED.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A




CLEANUP AND SEALING OF EDGE EROSION

B

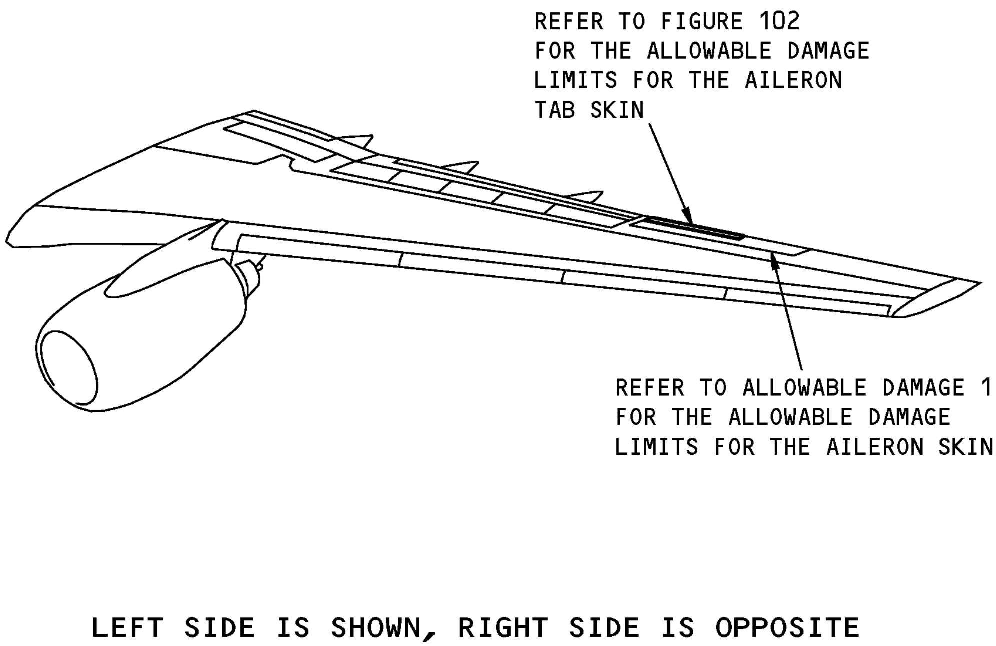
**Allowable Damage Limits**
**Figure 107**

## ALLOWABLE DAMAGE 2 - AILERON TAB SKIN

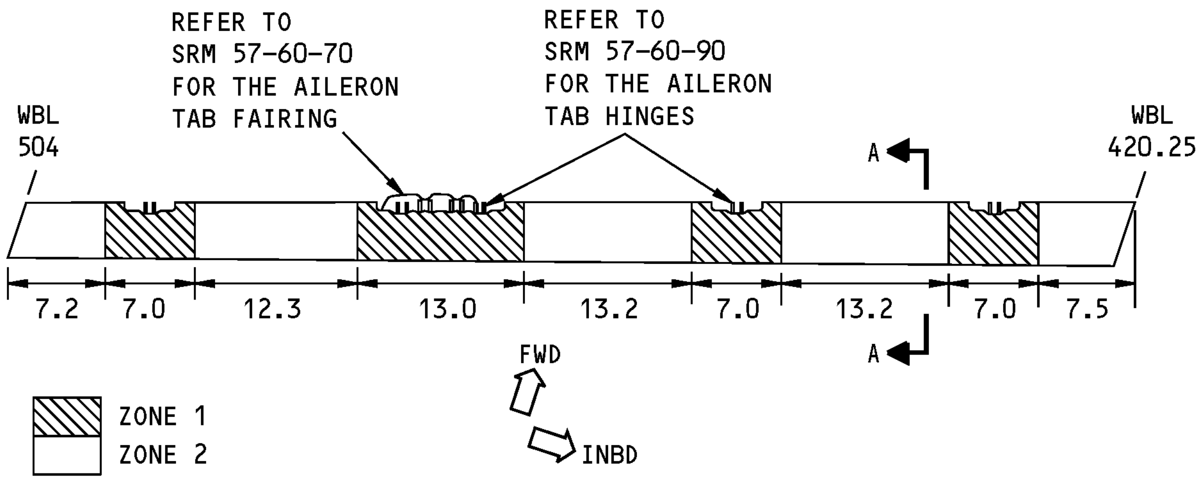
### 1. Applicability

A. This subject gives the allowable damage limits for the aileron tab skin as shown in Aileron Tab Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.

B. Refer to Aileron Tab Skin Zones, Figure 102/ALLOWABLE DAMAGE 2 for the aileron tab skin shapes.

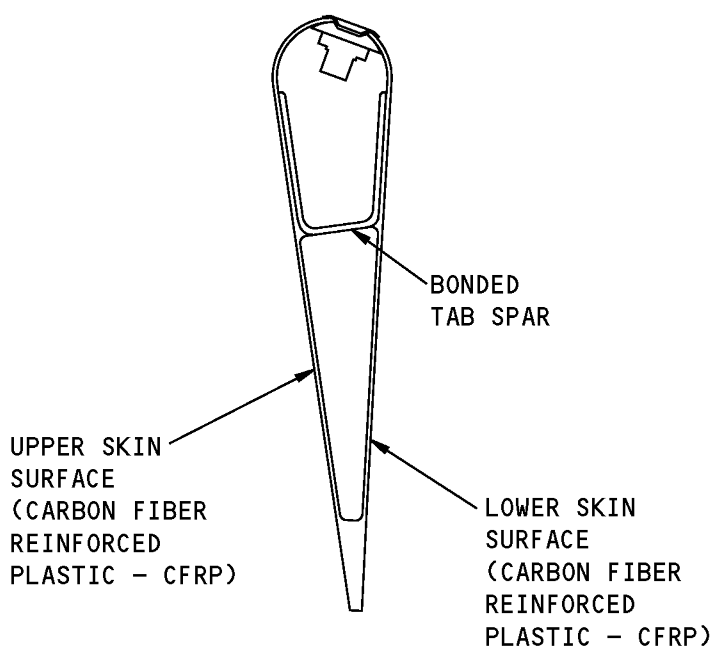



**Aileron Tab Skin Panel Location**
**Figure 101**




NOTE: ALL DIMENSIONS ARE IN INCHES.

UPPER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE




A-A

**Aileron Tab Skin Zones**
**Figure 102**

2. **General**

A. Do an inspection of the damaged area to find the length, width, and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C for the definitions of the length, width, and depth of damage.

B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two methods that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

(d) Repair the damage at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Use a vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04.

(2) Make a temporary seal with aluminum foil tape (speed tape).

(3) Keep a record of the location.

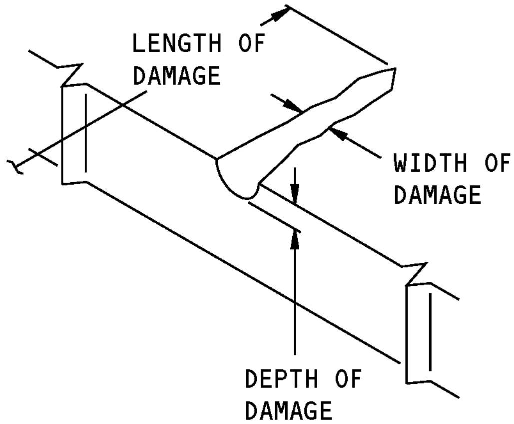
(4) Repair the damage at or before 400 flight hours from the time the seal was made.

E. The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause carbon fiber damage and are sealed.

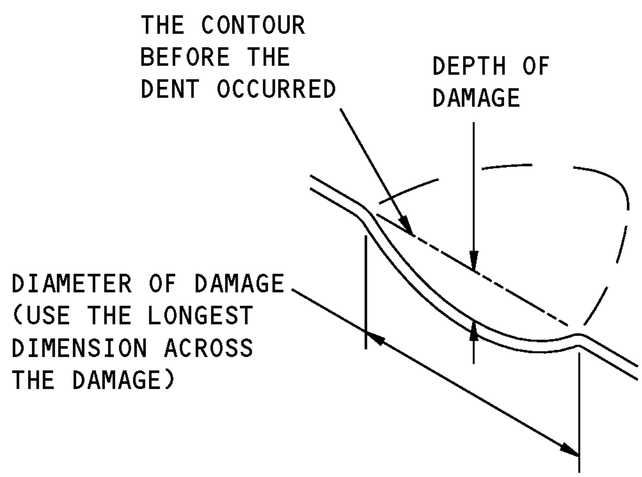
F. Make sure that the aileron tab is balanced. Refer to 51-60-03 for the balance procedures.

**Table 101**:

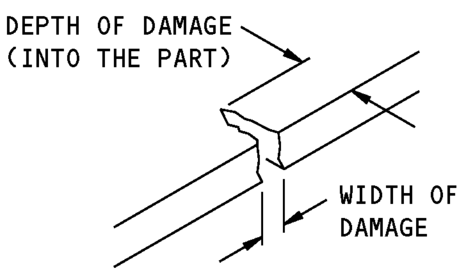
| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| **TYPE OF STRUCTURE** | **ZONE LOCATION** | **PARAGRAPH** |
| SOLID LAMINATE AREAS | 1 | 4.A |
| | 2 | 4.B |




DEFINITIONS FOR
NICK, GOUGE, OR SCRATCH DAMAGE
A




DEFINITIONS FOR
DENT DAMAGE
B




DEFINITIONS FOR
EDGE DAMAGE
C

**Damage Definitions**
**Figure 103**

**3. References**

| Reference | Title |
|---|---|
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-60-03 | AILERON TAB BALANCE PROCEDURE |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

**4. Allowable Damage Limits**

A. Zone 1 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if:

(a) The damage is a maximum of one ply

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) The damage is up to a maximum of 2.0 inches in length

(c) The damage is a maximum of 0.25 inch in width

(d) The damage is a minimum of 3.0 inches away from other damage.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if:

1) The damage is a maximum of 0.50 inch in diameter

2) The damage is a maximum of 0.50 inch in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch.

3) The damage is a minimum of 3.0 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) The damage is a maximum of 0.50 inch in diameter

(b) The damage is a minimum of 3.0 inches away from other damage.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail A if:

(a) The damage is a maximum of 0.50 inch in length

(b) The damage is a maximum of 0.50 inch in width.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

(b) A maximum of 0.25 inch in width

(c) A minimum of 0.50 inch away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail A or 4.0 inches.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail B .

B. Zone 2 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if:

(a) The damage is a maximum of one ply

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) The damage is up to a maximum of 1.50 inches in length

(c) The damage is a maximum of 0.25 inch in width

(d) The damage is a minimum of 3.0 inches away from other damage.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers is permitted if:

1) The damage is a maximum of 0.50 inch in diameter

2) The damage is a maximum of 0.50 inch in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch.

3) The damage is a minimum of 3.0 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) The damage is a maximum of 0.35 inch in diameter

(b) The damage is a minimum of 3.0 inches away from other damage.

(5) Delamination is permitted as shown in Figure 104/ALLOWABLE DAMAGE 2, Detail A if:

(a) The damage is a maximum of 0.40 inch in length

(b) The damage is a maximum of 0.40 inch in width.

(6) Edge damage is permitted if it is:

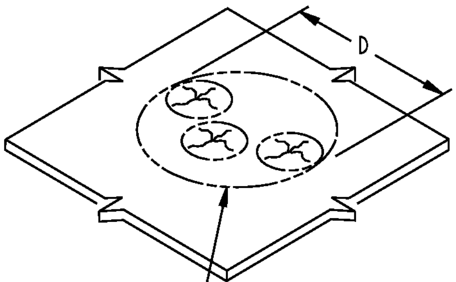
(a) A maximum of 0.08 inch in depth

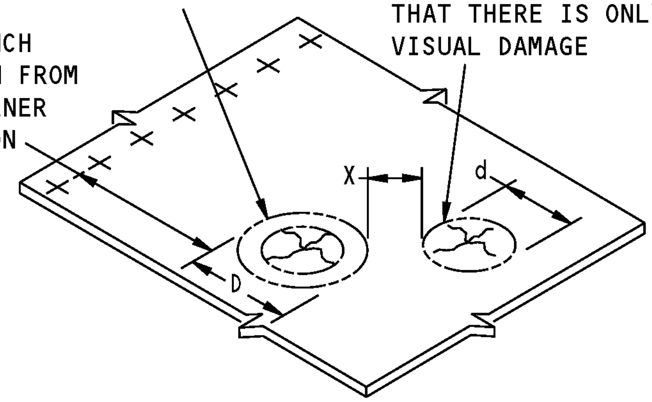
(b) A maximum of 0.25 inch in width

(c) A minimum of 0.50 inch away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail A .

(7) Edge Erosion is permitted as shown in Figure 104/ALLOWABLE DAMAGE 2, Detail B .




<u>NOTE</u>: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 1.0 INCH.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

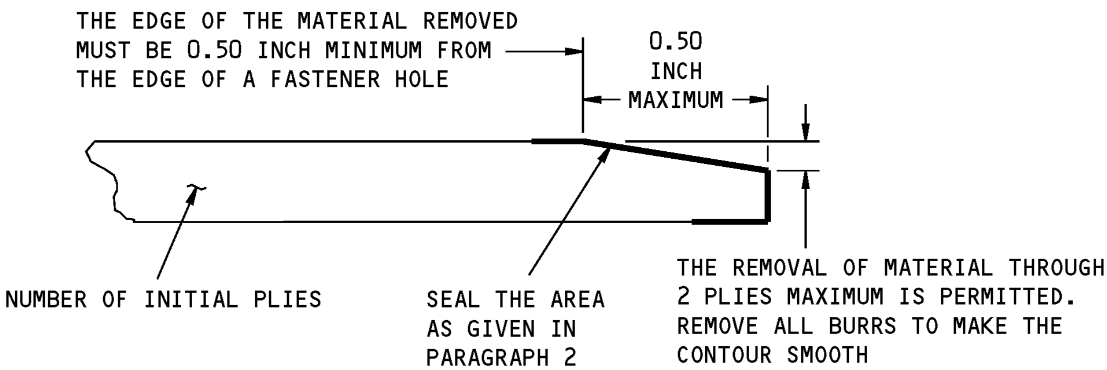
0.50 IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 2d.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A

**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 2)**




SEALING OF EROSION DAMAGE AT AN EDGE OF COMPOSITE PARTS

B

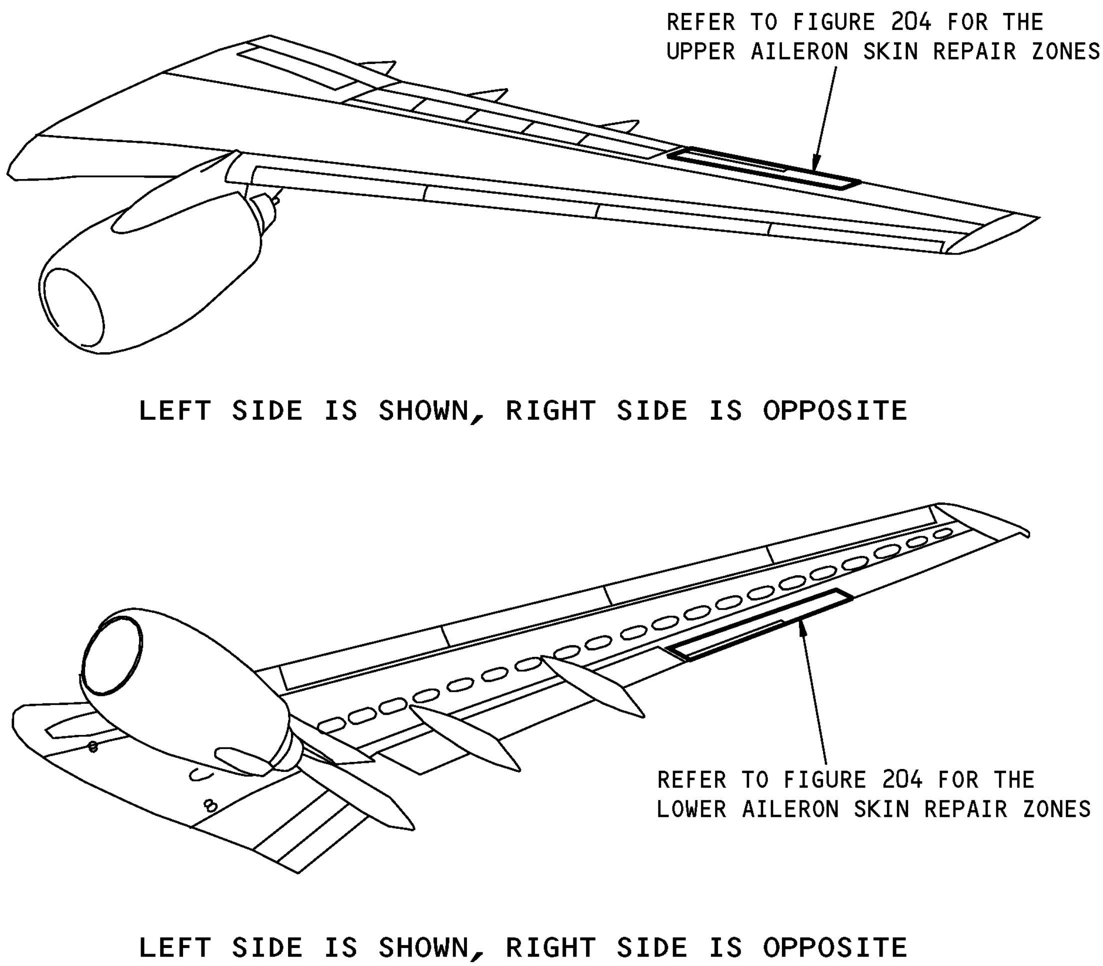
**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 2)**

## REPAIR 1 - AILERON SKIN

### 1. Applicability

A. Repair 1 is applicable to the upper and lower skin panels of the aileron shown in Wing Aileron Skin Panel Locations, Figure 201/REPAIR 1.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.




**Wing Aileron Skin Panel Locations**
**Figure 201**

### 2. General

A. Repair 1 gives instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the aileron, as necessary. Refer to AMM 27-11-11/401.

C. Remove the skin as necessary to get access to the inner surface. Refer to AMM 27-11-11/401.

(1) If the fastener hole is damaged, refer to 51-70-04 or 51-70-05 for a repair.

D. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) For the honeycomb core areas that have damage on a facesheet with 3 or less initial plies, the tap test is an alternative procedure to an instrumented NDT.

(2) For honeycomb areas that have damage on a facesheet with 4 or more initial plies, the tap test is an alternative procedure to an instrumented NDT if:

(a) It can be shown that a defect that is less than or equal to the maximum allowable damage size can be found.

(3) Refer to Damage Definitions, Figure 202/REPAIR 1 for the definitions of the diameter and depth of damage.

(4) Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

E. Do the repair as given in Paragraph 4./REPAIR 1

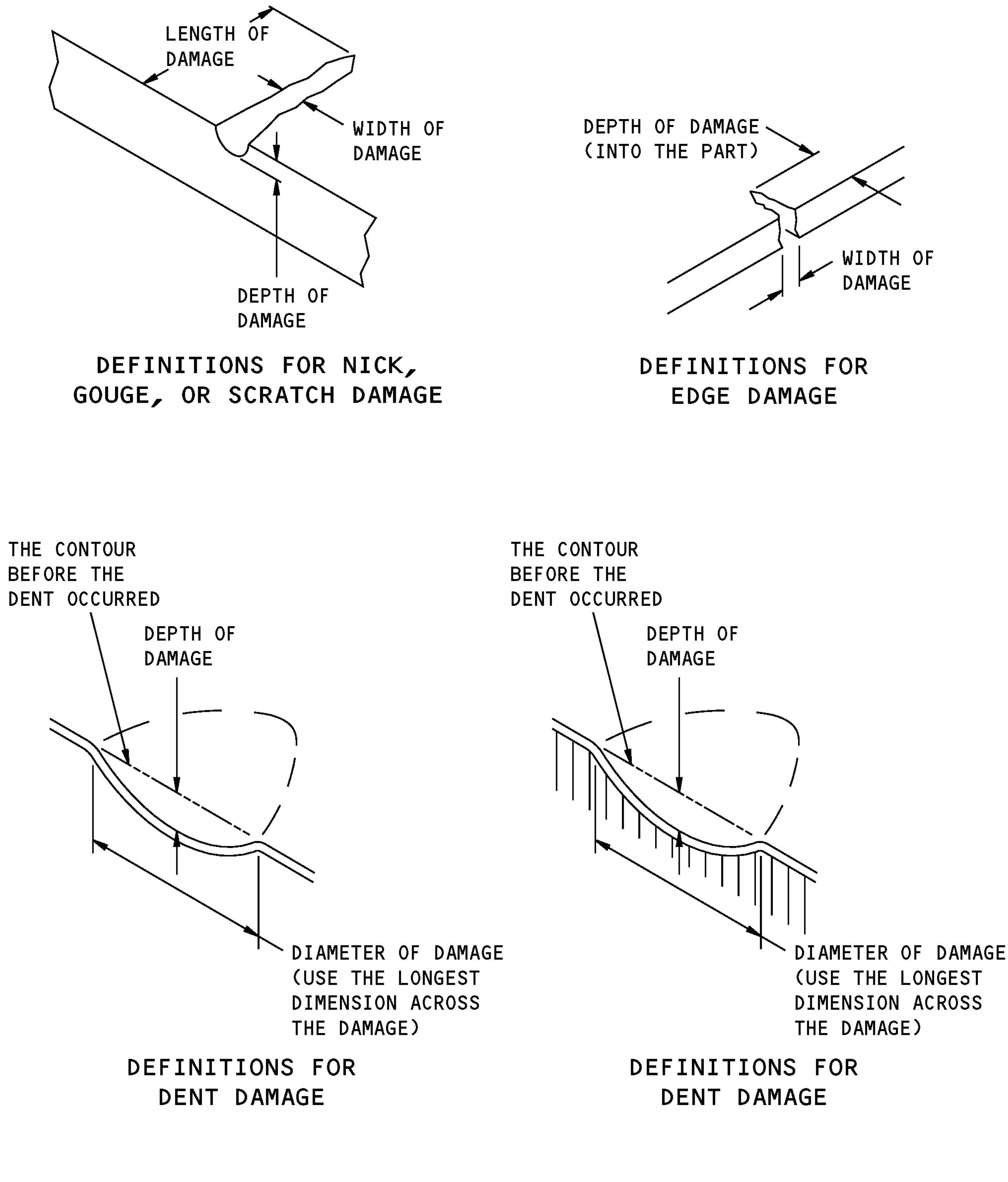
F. Install the skin you removed as necessary, to get access to the damage. Refer to AMM 27-11-11/401.

(1) If the fastener hole is damaged, refer to 51-70-04 or 51-70-05 for a repair.

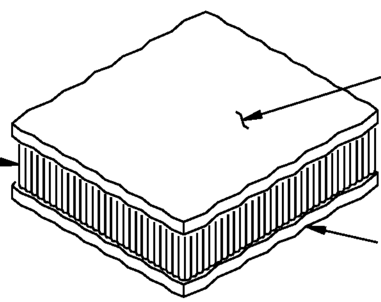
G. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

H. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

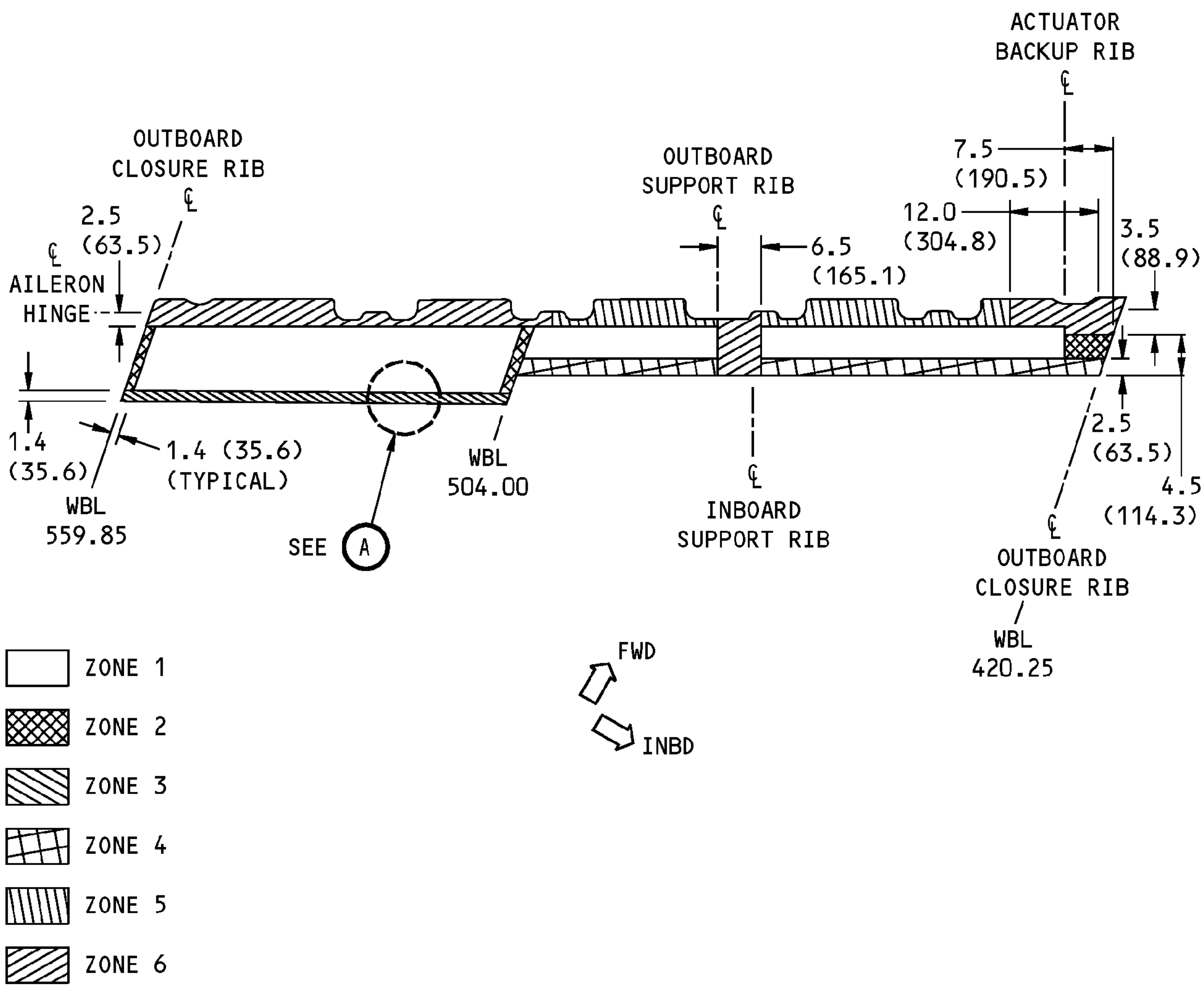
I. Install the aileron, as applicable. Refer to AMM 27-11-11/401.




**Damage Definitions**
**Figure 202**




**Definitions of the Facesheets**
**Figure 203**

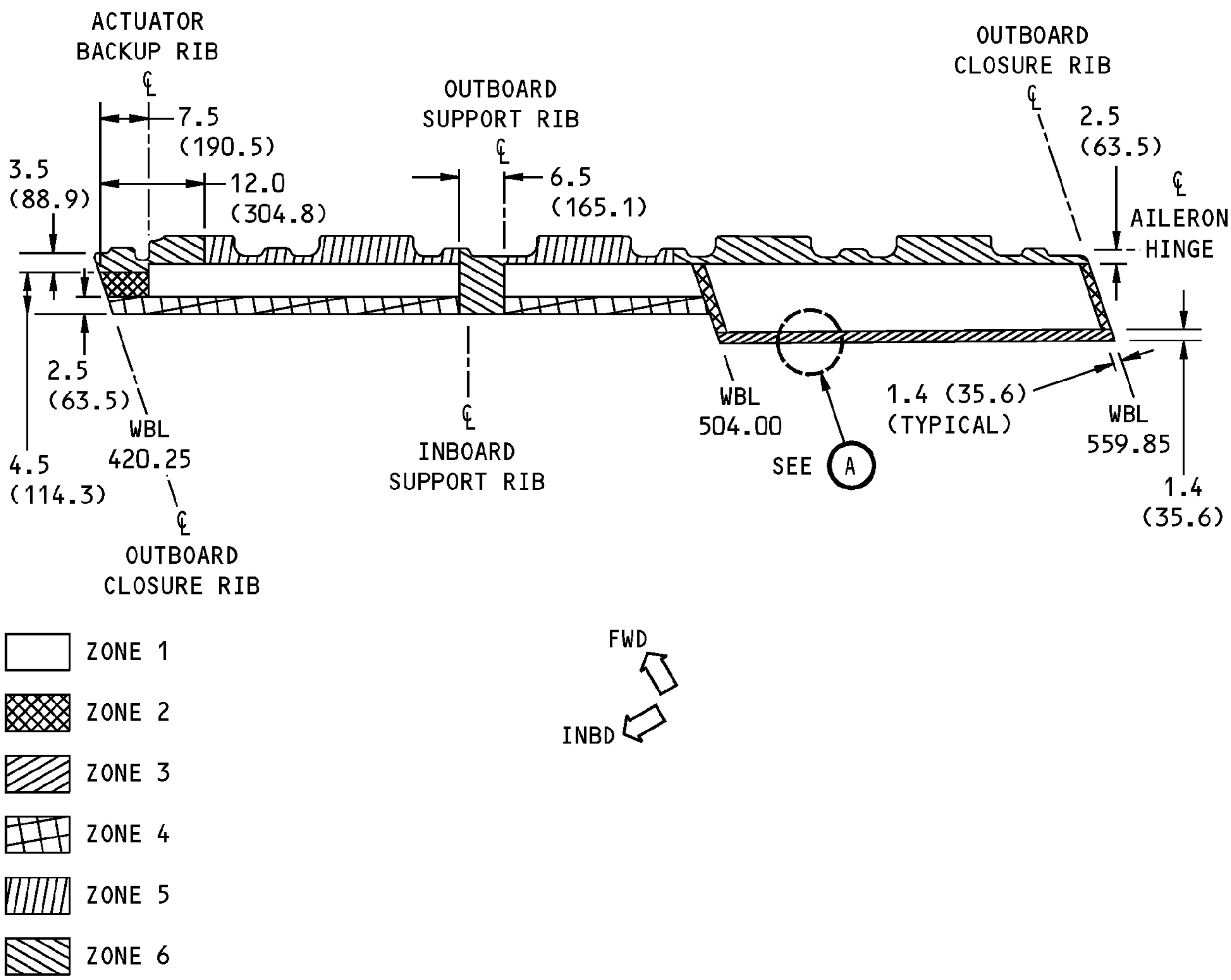



NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANELS

**Aileron Upper Skin Repair Zones**
**Figure 204 (Sheet 1 of 3)**

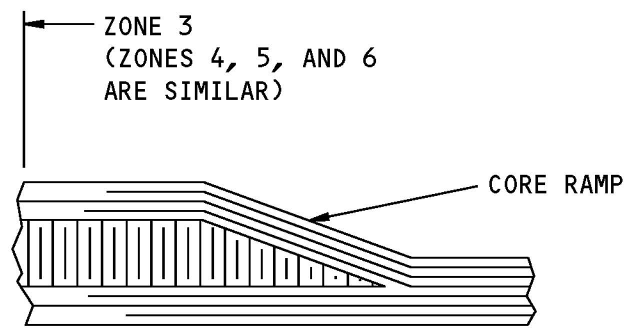



NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

RIGHT SIDE IS SHOWN, LEFT SIDE IS OPPOSITE

LOWER SKIN PANELS

**Aileron Upper Skin Repair Zones**
**Figure 204 (Sheet 2 of 3)**




DEFINITION OF THE SOLID LAMINATE
AND HONEYCOMB CORE AREAS
A

**Aileron Upper Skin Repair Zones**
**Figure 204 (Sheet 3 of 3)**

**3. References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-60-01 | AILERON BALANCING PROCEDURES |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-05 | REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS |
| AMM 51-21-00 | INTERIOR AND EXTERIOR FINISHES |
| AMM 27-11-11/401 | Aileron - Removal/Installation |
| 737 NDT Part 1, 51-01-01 | Inspection of Repairs to Composite Structure |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

**4. Repair Instructions**

A. If a dent is 2 inches in diameter or less and has no fiber damage or delamination, then fill the dent with potting compound and apply a fiberglass patch as given in Repair 14 of 51-70-04.

B. If Paragraph 4.A./REPAIR 1 is not applicable, then refer to Table 201/REPAIR 1 for the repair limits that are applicable to each zone.

**Table 201**:

| REPAIR INDEX | |
|---|---|
| ZONES | REPAIR TABLES |
| 1 | 202 |
| 2 and 3 | 203 |
| 4 and 5 | 204 |
| 6 | 205 |

C. If there is damage in more than one zone, then make sure:

(1) Each part of the repair that is in a given zone agrees with the repair size limits of that zone

(2) The type of repair and repair instructions agree with the minimum requirements for all the zones that the repair is in

(3) You refer to the example given in Paragraph 5./REPAIR 1 for damage in Zones 1 and 2.

D. For a repair made with wet layup materials, do as follows, as applicable:

(1) If the core is damaged, then fill with BMS 5-28, Type 7 potting compound.

(2) Use one repair ply of fabric for each initial ply that was damaged.

(3) Add two structural fiberglass plies of fabric for each facesheet that is repaired. Refer to 51-70-04. Put one structural ply at ±45 degrees to the core ribbon direction and the other at 0 or 90 degrees. Add a third nonstructural ply made of fiberglass where the ply direction is optional.

**NOTE**: Repair plies or added plies are not necessary in the repair of a delamination at an edge if the delamination is a minimum of 2D (D = fastener diameter) away from a fastener hole.

(4) Inspect a Category B repair after each 600 flight hour interval or more frequently. Refer to 737 NDT Part 1, 51-01-01 and 51-70-04 for the inspection procedures. If deterioration is found, then the repair must be replaced with a Category A repair.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator, can be used.

E. For a repair made with preimpregnated layup materials, use the same number of repair plies as the number of initial plies that were damaged.

F. Use the repair instructions that follow for a repair in Zone 1 made with wet layup materials at 200°F (93°C) cure.

(1) Repair the damage as given in 51-70-04, but for each facesheet (over a honeycomb core or solid laminate) that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that are removed. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definition of a facesheet.

(b) Add one ply (structural) that is + or - 45 degrees.

(c) Add a second ply (structural) that is 0 or 90 degrees.

(d) Add a third ply (non-structural) made of fiberglass with an optional ply direction.

G. Use the repair instructions that follow for a repair in Zone 1 made with preimpregnated layup materials at 250°F (121°C) cure.

(1) Repair the damage as given in 51-70-05, but for each facesheet (over a honeycomb core or solid laminate) that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that are removed from a facesheet. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definition of a facesheet.

(b) Add one ply (structural) that is + or - 45 degrees.

(c) Add a second ply (structural) that is 0 or 90 degrees.

(d) Add a third ply (structural) that is + or - 45 degrees.

(e) Add a fourth ply (non-structural) made of fiberglass with an optional ply direction.

H. Use the repair instructions that follow for a repair in Zones 1, 2, and 3 made with preimpregnated layup materials at 350°F (177°C) cure.

(1) Repair the damage as given in 51-70-05, but for each facesheet (over a honeycomb core) or solid laminate area that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that are removed from a facesheet or solid laminate area. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definition of a facesheet.

(b) Add one ply (nonstructural) made of fiberglass with an optional ply direction.

**5. Example of Repair Options for Damage That Is In Zones 1 and 2.**

A. Refer to Table 201/REPAIR 1 for the repairs that are applicable in Zone 1.

(1) 250°F preimpregnated layup repairs are permitted if:

(a) The damage is a maximum of 10.0 inches in length

(b) The damage is a maximum of 65 percent of the smallest dimension across the panel at the damage location

(c) You add two more structural plies than the initial number of structural plies and one nonstructural fiberglass ply

(d) You then add a nonstructural ply made of fiberglass.

(2) 200°F wet layup repairs are permitted if:

(a) The damage is a maximum of 10.0 inches in length

(b) The damage is a maximum of 65 percent of the smallest dimension across the panel at the damage location

(c) You add two more structural plies than the initial number of structural plies

(d) You then add a nonstructural ply made of fiberglass.

(3) 350°F preimpregnated layup repairs are permitted with no size limits.

B. Refer to Table 201/REPAIR 1 for the repairs that are applicable in Zone 2.

(1) 200°F wet layup repairs are permitted if:

(a) The damage is a maximum of 10.0 inches in length

(b) The damage is a maximum of 65 percent of the smallest dimension across the panel at the damage location

(c) You add four more structural plies than the initial number of structural plies

(d) You then add a ply nonstructural made of fiberglass.

(2) 250°F preimpregnated layup repairs are permitted since they are listed in the table.

(3) 350°F preimpregnated layup repairs are permitted with no size limits.

C. Do a repair that agrees with the requirements of both Zones 1 and 2.

(1) Since a 250°F preimpregnated layup repair is not permitted in Zone 2, it can not be used for damage that is in both Zones 1 and 2.

(2) A 350°F preimpregnated layup repair is permitted for both Zones 1 and 2 with no size limits and the same repair instructions (Paragraph 4.F./REPAIR 1).

(3) A 200°F wet layup repair is permitted for both Zones 1 and 2 if the conditions that follow are true:

(a) The part of the damage that is in Zone 1 must agree with the size limits given in Paragraphs A.(1)(a) and A.(1)(b).

(b) The part of the damage that is in Zone 2 must agree with the size limits given in Paragraphs B.(1)(a) and B.(1)(b).

(c) Zone 1 adds two more structural plies than the initial number of structural plies and Zone 2 adds four more. The result is that you must use four more structural plies in both zones to agree with the minimum requirements of Zone 1. A nonstructural ply made of fiberglass is then added.

Table 202:

| REPAIR DATA FOR ZONE 1 OF THE 350°F (177°C) CURE, AILERON HONEYCOMB SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | Damage that is a maximum of:<br>- 2.00 inch in diameter<br>- 30 percent of the smallest dimension across the panel at the damage location<br>- one facesheet and the honeycomb core in depth<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 10.0 inches in diameter<br>- 65 percent of the smallest dimension across the panel at the damage location<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 10.0 inches in diameter<br>- 65 percent of the smallest dimension across the panel at the damage location<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | There are no limits on the dimension of the repair. |
| REPAIR PROCEDURES | SRM 51-70-04 and Paragraph 4.D | SRM 51-70-04 and Paragraph 4.F | SRM 51-70-05 and Paragraph 4.G | SRM 51-70-05 and Paragraph 4.H |

Table 203:

| REPAIR DATA FOR ZONE 2 AND 3 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED IAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS FOR ZONE 2 | Damage that is a maximum of:<br>- 2.00 inches in diameter<br>- 30 percent of the smallest dimension across the panel at the damage location<br>- one facesheet depth<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 10.0 inches in diameter<br>- 65 percent of the smallest dimension across the panel at the damage location<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 10.0 inches in diameter<br>- 65 percent of the smallest dimension across the panel at the damage location<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | There are no size limits on the repair. |

| REPAIR DATA FOR ZONE 2 AND 3 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED IAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS FOR ZONE 3 | - Contact The Boeing Company | - Contact The Boeing Company | - Contact The Boeing Company | There are no size limits on the repair. |
| REPAIR PROCEDURES | SRM 51-70-04 and Paragraph 4.D | SRM 51-70-04 and Paragraph 4.F | SRM 51-70-05 and Paragraph 4.G | SRM 51-70-05 and Paragraph 4.H |

**Table 204**:

| REPAIR DATA FOR ZONE 4 AND 5 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED IAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS FOR ZONE 5 | A repair is not permitted. | Damage that is a maximum of:<br>- 2.0 inches in diameter<br>- 10 percent of the edgeband length on the side of the damage, as applicable<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 2.0 inches in diameter<br>- 10 percent of the edgeband length on the side of the damage, as applicable<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | There are no size limits on the repair. |
| REPAIR SIZE AND LIMITS FOR ZONE 4 | - Contact The Boeing Company | - Contact The Boeing Company | - Contact The Boeing Company | There are no size limits on the repair. |
| REPAIR PROCEDURES | ———— | SRM 51-70-04 and Paragraph 4.F | SRM 51-70-05 and Paragraph 4.G | SRM 51-70-05 and Paragraph 4.H |

**Table 205**:

| REPAIR DATA FOR ZONE 6 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED IAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | A repair is not permitted. | Damage that is a maximum of:<br>- 2.0 inches in diameter | Damage that is a maximum of:<br>- 2.0 inches in diameter | There are no size limits on the repair. |

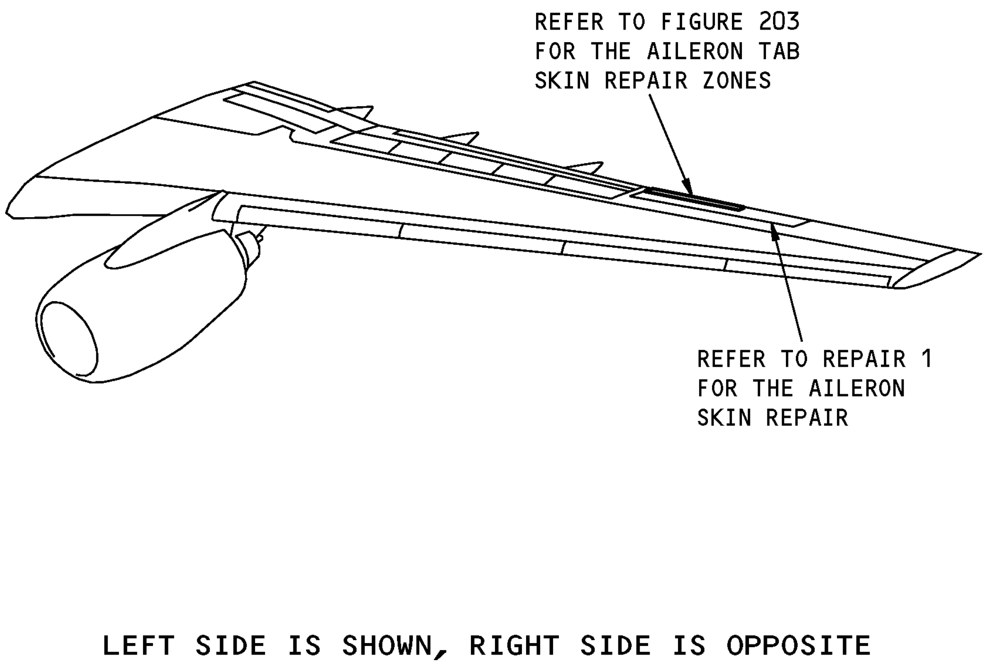
| REPAIR DATA FOR ZONE 6 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED IAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| | | - 10 percent of the edgeband length on the side of the damage, as applicable<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | - 10 percent of the edgeband length on the side of the damage, as applicable<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | |
| REPAIR PROCEDURES | —— | SRM 51-70-04 and Paragraph 4.F | SRM 51-70-05 and Paragraph 4.G | SRM 51-70-05 and Paragraph 4.H |

## REPAIR 2 - AILERON TAB SKIN

### 1. Applicability

A. Repair 2 is applicable to the Aileron Tab Skin shown in Aileron Tab Skin Panel Location, Figure 201/REPAIR 2.

B. Repair 2 is applicable to damage that is more than the limits permitted in Allowable Damage 2. Refer to Allowable Damage 2 for the type and size of damage that is permitted.




**Aileron Tab Skin Panel Location**
**Figure 201**

### 2. General

A. Repair 2 gives instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the aileron, as necessary. Refer to AMM 27-11-11/401.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

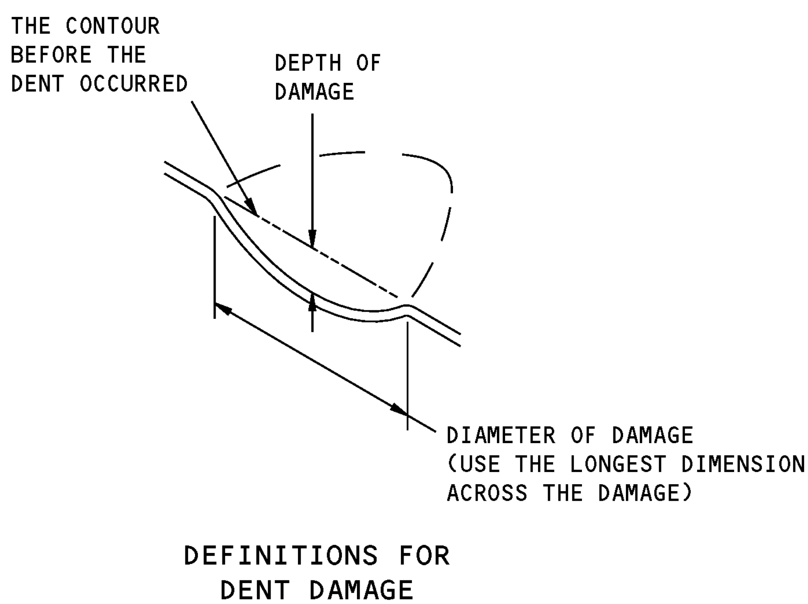
(1) Refer to Definitions of Damage Size, Figure 202/REPAIR 2 for the definitions of the diameter and depth of damage.

D. Do the repair as given in Paragraph 4./REPAIR 2

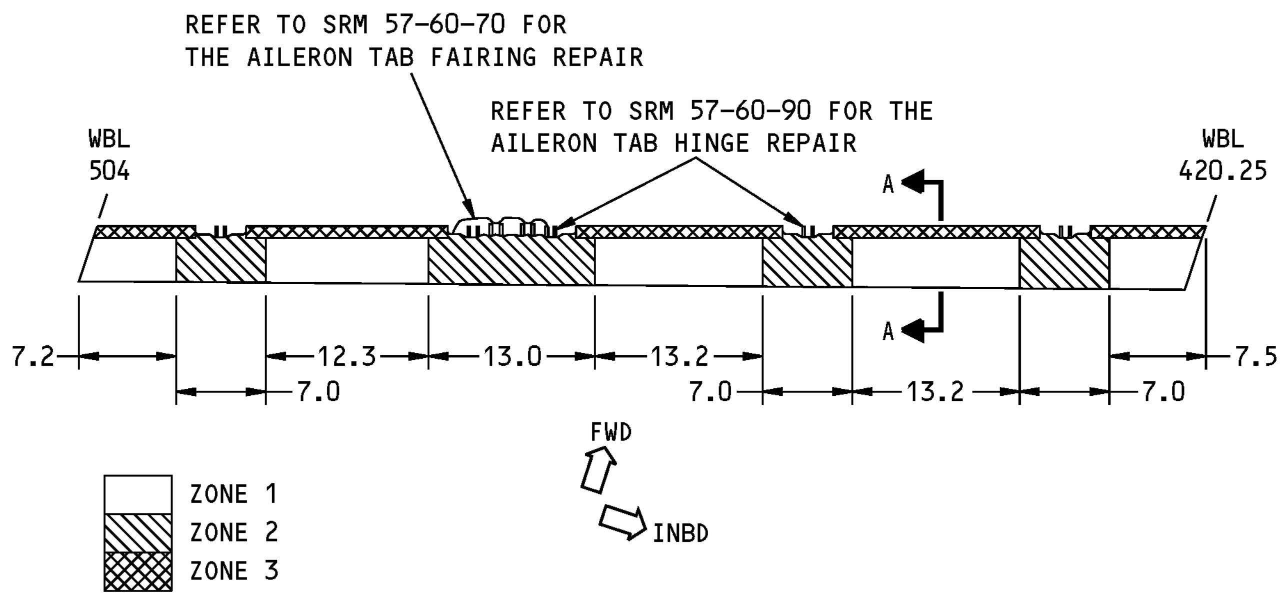
E. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

F. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

G. Install the aileron tab, as applicable. Refer to AMM 27-11-11/401.

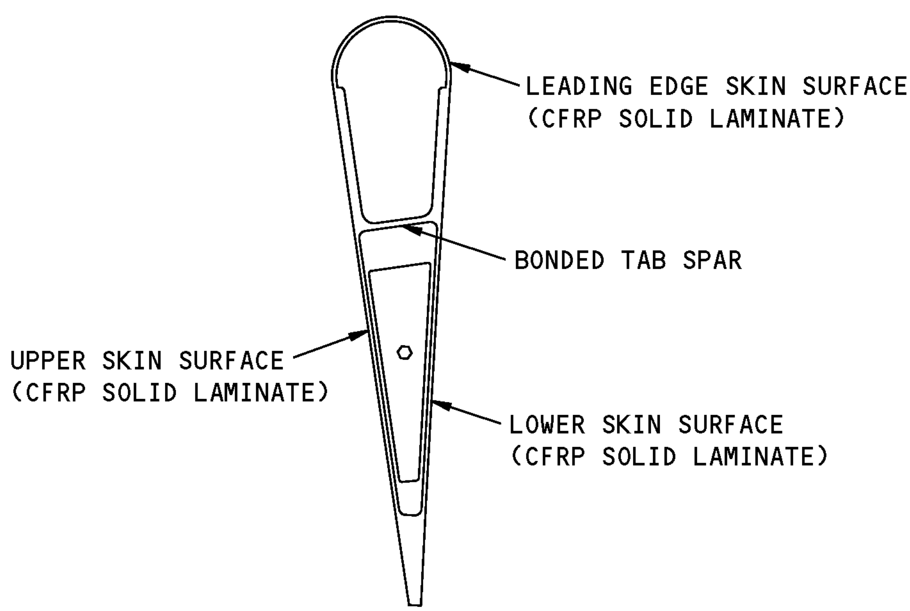



**Definitions of Damage Size**
**Figure 202**




NOTE: ALL DIMENSIONS ARE IN INCHES.

UPPER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE




A-A

**Aileron Tab Skin Repair Zones**
**Figure 203**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-60-01 | AILERON BALANCING PROCEDURES |
| 51-60-03 | AILERON TAB BALANCE PROCEDURE |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| AMM 51-21-00 | INTERIOR AND EXTERIOR FINISHES |
| AMM 27-11-11/401 | Aileron - Removal/Installation |
| 737 NDT Part 1, 51-01-01 | Inspection of Repairs to Composite Structure |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

## 4. Repair Instructions

A. If a dent is 2 inches in diameter or less, and has no fiber damage or delamination, then fill the dent with potting compound and apply a fiberglass patch as given in Repair 14 of 51-70-04.

B. If Paragraph 4.A./REPAIR 2 is not applicable, then refer to:

(1) Table 201/REPAIR 2 for the repair data that is applicable to damage in Zone 1

(2) Table 202/REPAIR 2 for the repair data that is applicable to damage in Zone 2

(3) Table 203/REPAIR 2 for the repair data that is applicable to damage in Zone 3.

C. For repairs made with wet layup materials, do as follows, as applicable:

(1) Use one repair ply of fabric for each initial ply that was damaged.

(2) Add two structural plies of fabric for each facesheet that is repaired. Put one structural ply at ±45 degrees to the core ribbon direction and the other at 0 or 90 degrees.

**NOTE**: Repair plies or added plies are not necessary in the repair of delamination at an edge if the delamination is a minimum of 2D (D = fastener diameter) away from a fastener hole.

(3) Inspect Category B repairs after each 800 flight hour interval or more frequently. Refer to 737 NDT Part 1, 51-01-01 for inspection procedures. If deterioration is found, then they must be replaced with Category A repairs.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

D. For repairs made with preimpregnated layup materials, use the same number of repair plies as the number of initial plies that were damaged.

Table 201:

| REPAIR DATA FOR ZONE 1 OF THE 350°F (177°C) CURE, AILERON TAB LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | Damage that is a maximum of:<br>- 1.0 inch in diameter<br>- 30 percent of the smallest dimension across the panel.<br>- One facesheet in depth<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 2.0 inches in diameter<br>- 50 percent of the smallest dimension across the panel.<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 2.50 inches in diameter<br>- 50 percent of the smallest dimension across the panel. | There are no size limits on the repair. |
| REPAIR PROCEDURES | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-05 and Paragraph 4.D | SRM 51-70-05 and Paragraph 4.D |

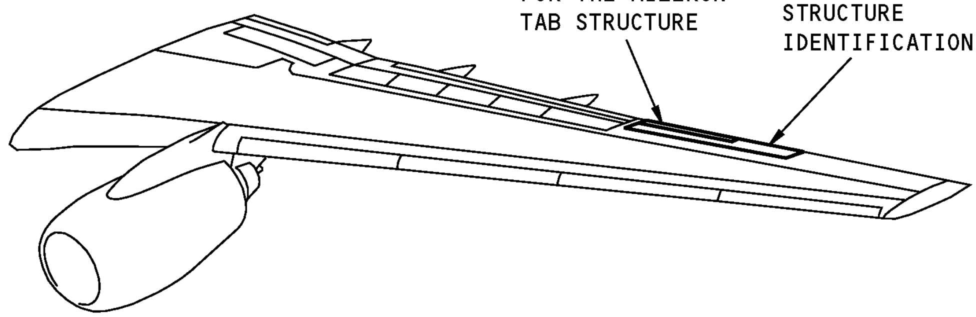
Table 202:

| REPAIR DATA FOR ZONE 2 OF THE 350°F (177°C) CURE, AILERON TAB LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | Damage that is a maximum of:<br>- 0.50 inch in diameter<br>- 15 percent of the edgeband length on the side of the damage, as applicable<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 1.0 inch in diameter<br>- 10 percent of the edgeband length on the side of the damage, as applicable<br>One repair for each 144 square inches<br>3.0 inches minimum clearance from:<br>- other repairs<br>- fastener holes<br>- panel edges | Damage that is a maximum of:<br>- 1.5 inches in diameter<br>- 10 percent of the edgeband length on the side of the damage, as applicable | There are no size limits on the repair |
| REPAIR PROCEDURES | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-04 and Paragraph 4.C | SRM 51-70-05 and Paragraph 4.D | SRM 51-70-05 and Paragraph 4.D |

Table 203:

| REPAIR DATA FOR ZONE 3 OF THE 350°F (177°C) CURE, AILERON TAB LAMINATE SKIN PANELS | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | Damage that is a maximum of:<br>- 1.0 inch in length<br>- 30 percent of the length of the leading edge between cutouts | Damage that is a maximum of:<br>- 2.0 inches in length<br>- 50 percent of the length of the leading edge between cutouts | Damage that is a maximum of:<br>- 2.5 inches length<br>- 50 percent of the length of the leading edge between cutouts | There are no size limits on the repair. |
| REPAIR PROCEDURES | SRM 51-70-04 | SRM 51-70-04 | SRM 51-70-05 | SRM 51-70-05 |

## IDENTIFICATION 1 - AILERON STRUCTURE



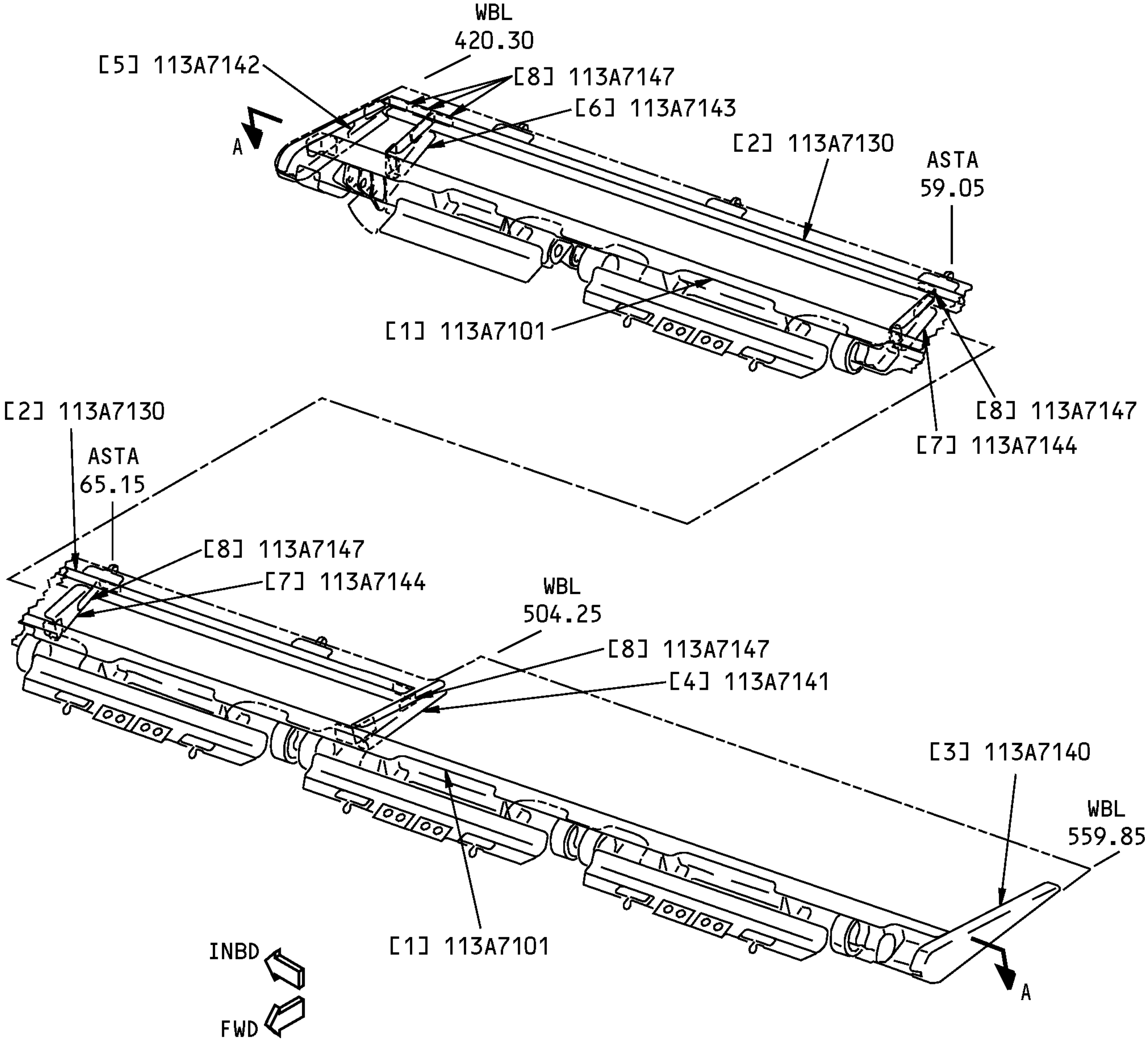

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT AILERON IS SHOWN, RIGHT AILERON IS OPPOSITE

**Aileron Structure Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A7001 | Aileron Installation |
| 113A7002 | Aileron/Tab Assembly |
| 113A7100 | Aileron Assembly |
| 113A7101 | Front Spar Assembly - Aileron |



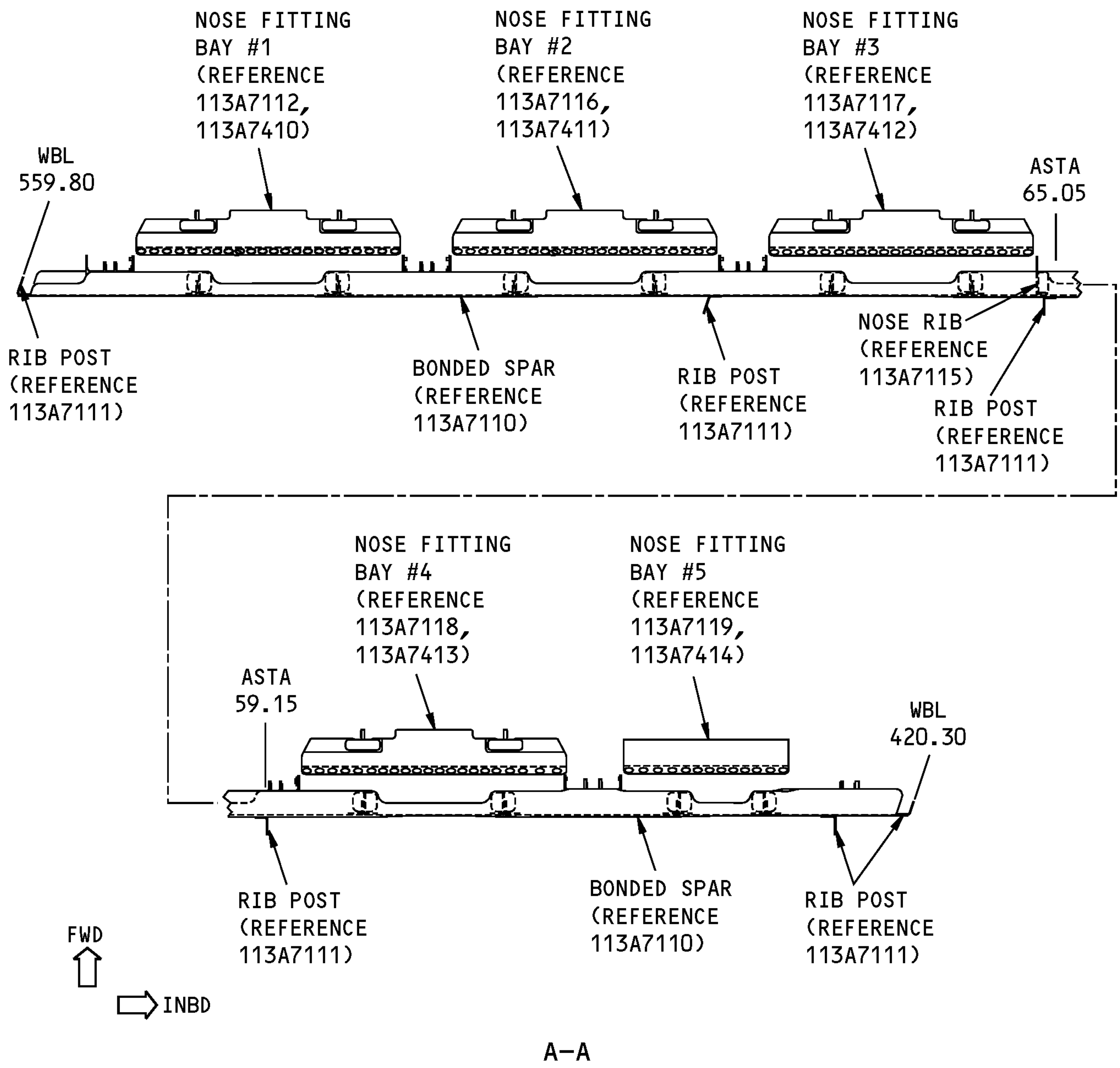

LEFT AILERON IS SHOWN, RIGHT AILERON IS OPPOSITE

AILERON ASSEMBLY

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

**Aileron Structure Identification**
**Figure 2 (Sheet 1 of 2)**

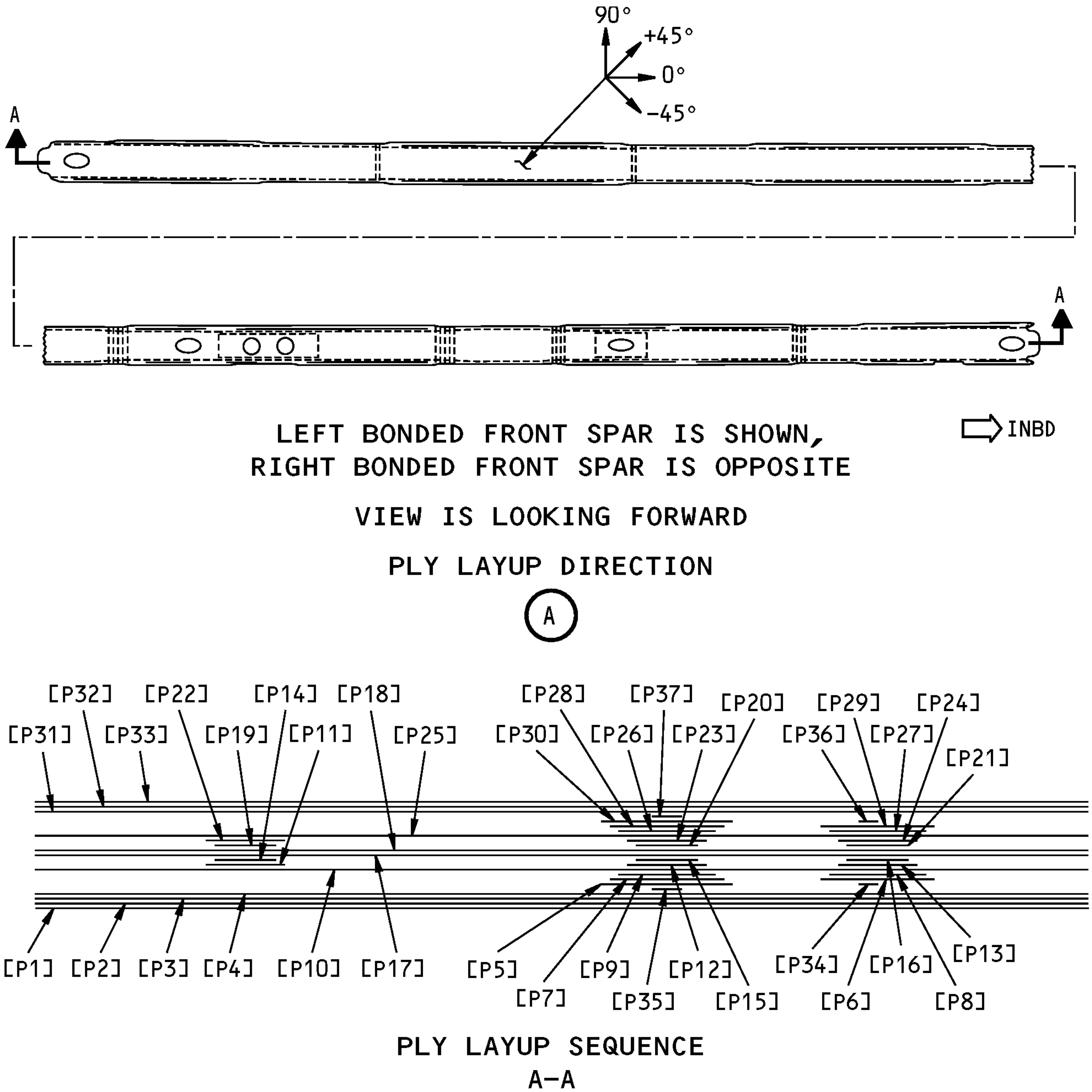



**Aileron Structure Identification**
**Figure 2 (Sheet 2 of 2)**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Bonded Front Spar Assembly | | | |
| | Bonded Spar | | Carbon Fiber Reinforced Plastic (CFRP) woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 3 | |
| | Bonded Rib Post (6) | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to the production drawing for each rib post ply layup. Refer to Figure 4 for a typical rib post ply layup | |
| | Nose Fitting (10) | | 15-5PH Investment Casting | |
| | Nose Rib (5) | 0.250 (6.35) | 7075-T7451 plate | |
| [2] | Bonded Rear Spar | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 5 | |
| [3] | Bonded Outboard Closure Rib | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 6 | |
| [4] | Bonded Tab Cutout Rib | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 7 | |
| [5] | Bonded Inboard Closure Rib | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 8 | |
| [6] | Bonded Actuator Backup Rib | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 9 | |
| [7] | Bonded Support Rib | | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 10 | |
| [8] | Rib Post | 0.050 (1.27) | Ti-6Al-4V titanium sheet as given in MIL-T-9046, Code AB-1 in the annealed condition | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTES

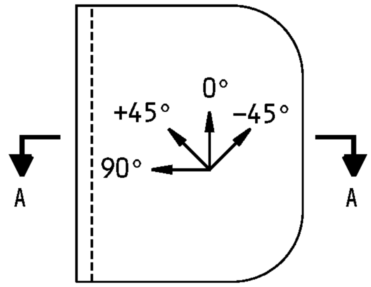
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

**Ply Ribbon Direction and Ply Sequence for the Bonded Spar,Figure 2, Item [1]**
**Figure 3**

**Table 3**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 3 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1 | Optional | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class III, Style 108 |
| P2, P4, P34, P5, P8, P9, P11 thru P13, P17, P18, P22 thru P24, P26, P27, P30, P36, P31, P33 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P3, P35, P6, P7, P10, P14 thru P16, P19 thru P21, P25, P28, P29, P37, P32 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

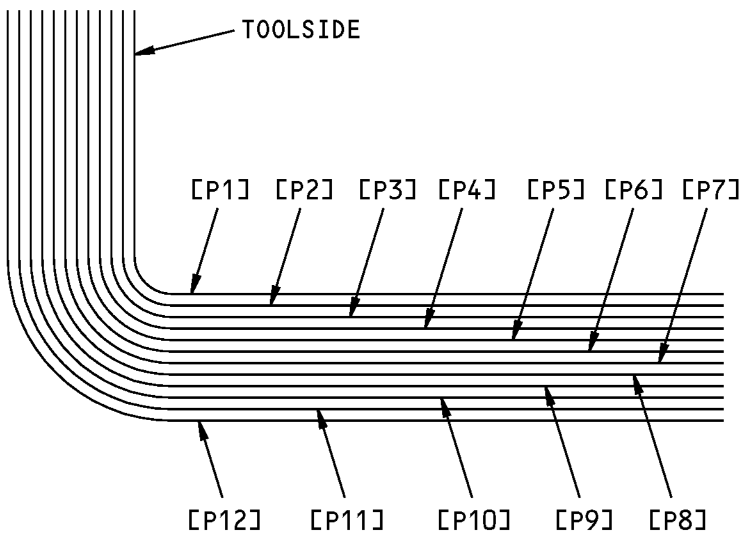



PLAN VIEW

BONDED RIB POST

PLY LAYUP DIRECTION

A




PLY LAYUP SEQUENCE
A-A

NOTES

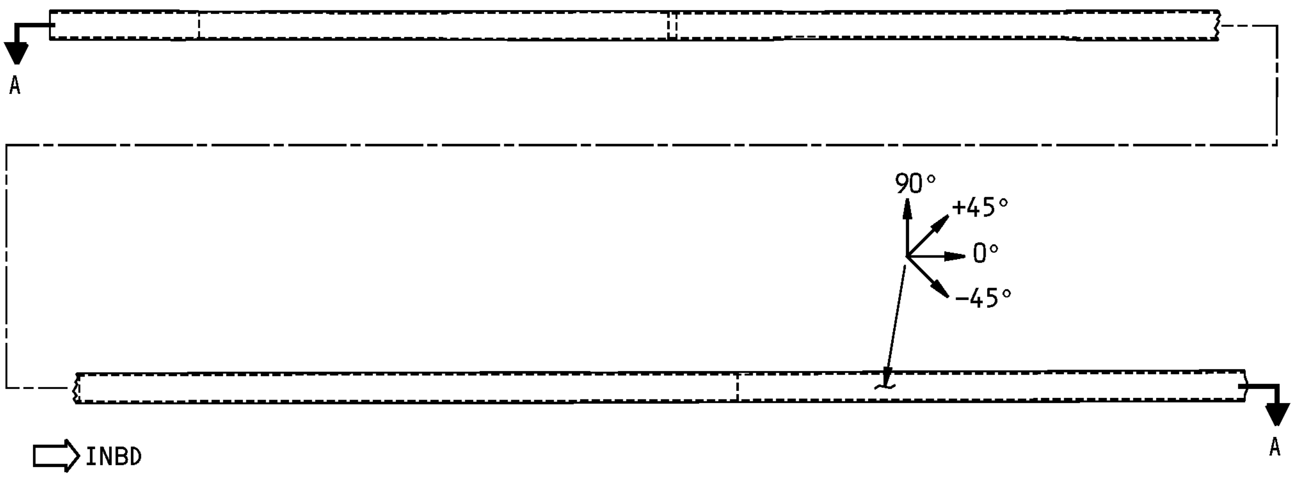
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

**Ply Ribbon Direction and Ply Sequence for the Bonded Rib Post, Figure 2, Item [1]**
**Figure 4**

Table 4:

| PLY MATERIAL AND DIRECTION FOR FIGURE 4 | | |
|---|---|---|
| PLY | DIRECTION | MATERIAL |
| P1, P3, P5, P8, P10, P12 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P4, P6, P7, P9, P11 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

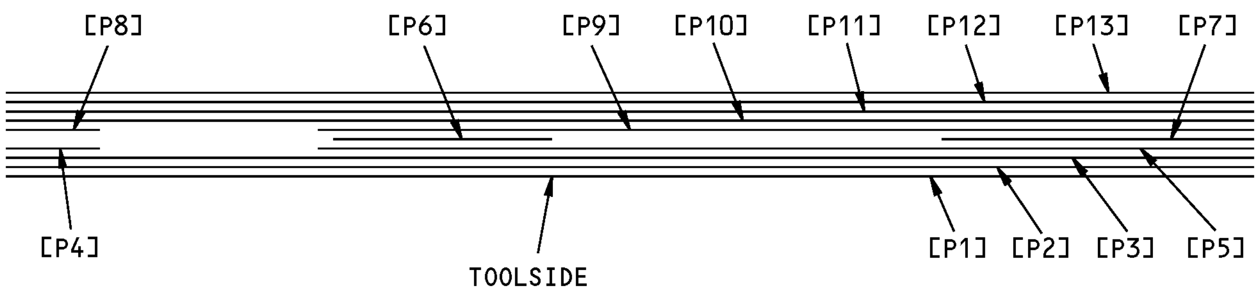



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

REAR VIEW OF BONDED REAR SPAR

PLY LAYUP DIRECTION

A




PLY LAYUP SEQUENCE
A-A

NOTES

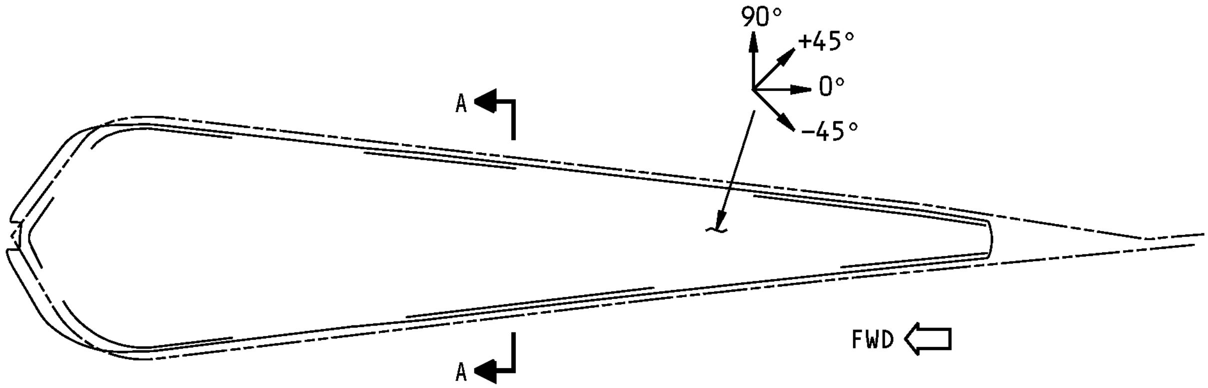
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

**Ply Direction and Ply Sequence for the Bonded Rear Spar, Figure 2, Item [2]**
**Figure 5**

**Table 5**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 5 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P6, P7, P10, P12 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P4, P5, P8, P9, P11 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P13 | Optional | Epoxy impregnated glass woven fabric as given in BMS 8-139, Class III, Style 108 |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

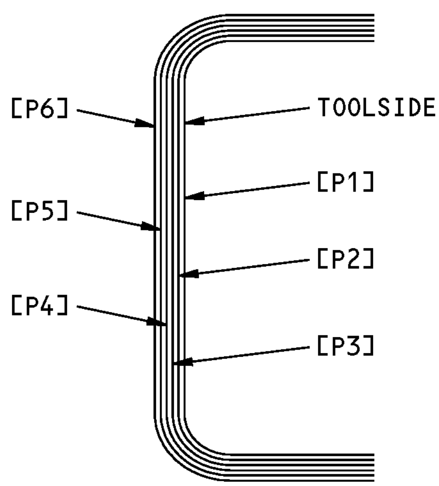



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED OUTBOARD CLOSURE RIB

PLY LAYUP DIRECTION

A




PLY LAYUP SEQUENCE
A-A

NOTES

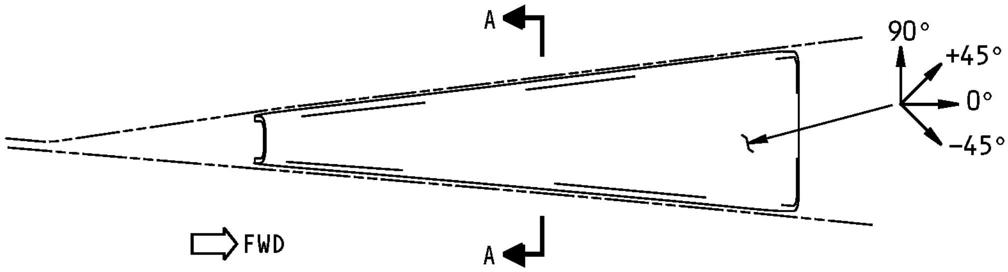
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

**Ply Direction and Ply Sequence for the Bonded Outboard Closure Rib, Figure 2, Item [3]**
**Figure 6**

**Table 6**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 6 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P4, P6 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P5 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

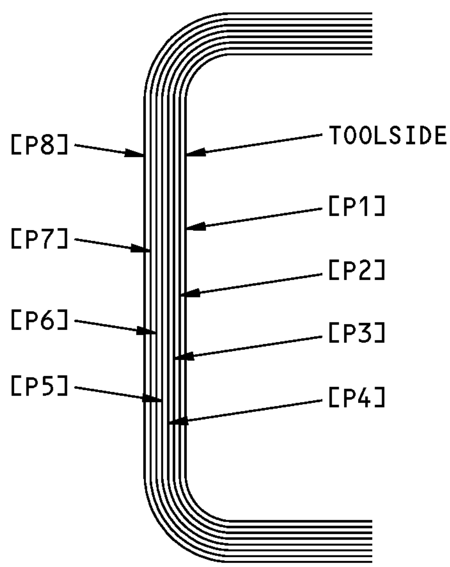



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED TAB CUTOUT RIB

PLY LAYUP DIRECTION

A




PLY LAYUP SEQUENCE
A-A

NOTES

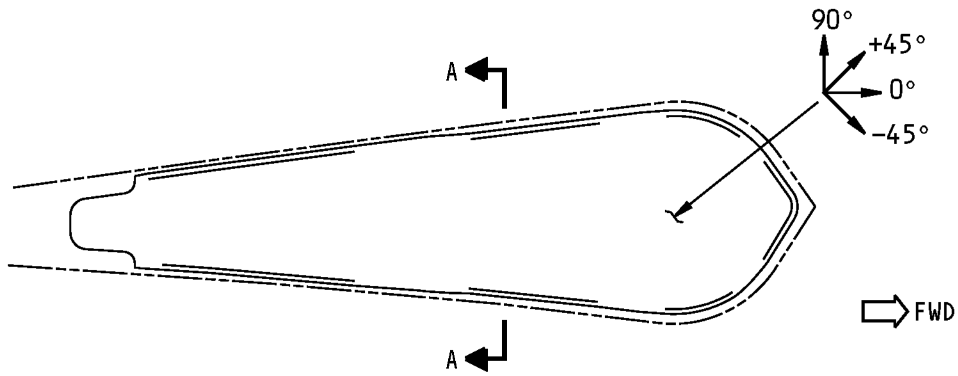
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

**Ply Direction and Ply Sequence for the Tab Cutout Rib, Figure 2, Item [4]**
**Figure 7**

**Table 7**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 7 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P6, P8 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P4, P5, P7 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

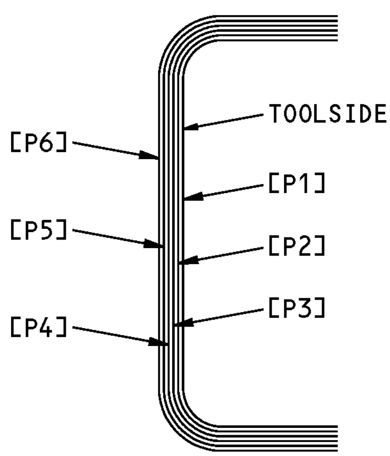



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED INBOARD CLOSURE RIB

PLY LAYUP DIRECTION

A




PLY LAYUP SEQUENCE
A-A

NOTES

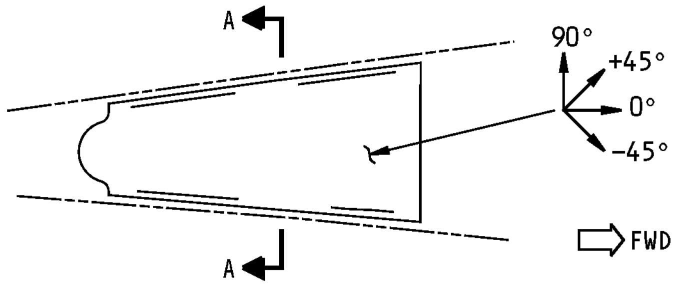
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

**Ply Direction and Ply Sequence for the Inboard Closure Rib, Figure 2, Item [5]**
**Figure 8**

**Table 8**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 8 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P4, P6 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P5 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

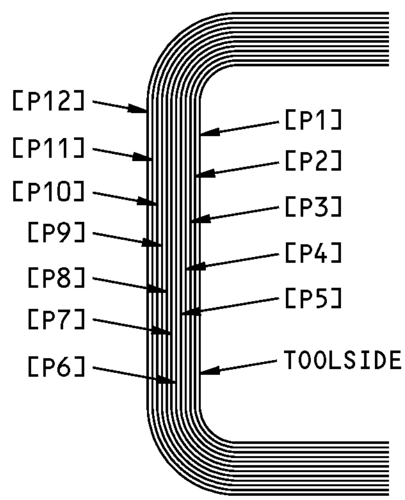



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED ACTUATOR BACKUP RIB

PLY LAYUP DIRECTION

A




PLY LAYUP SEQUENCE
A-A

NOTES

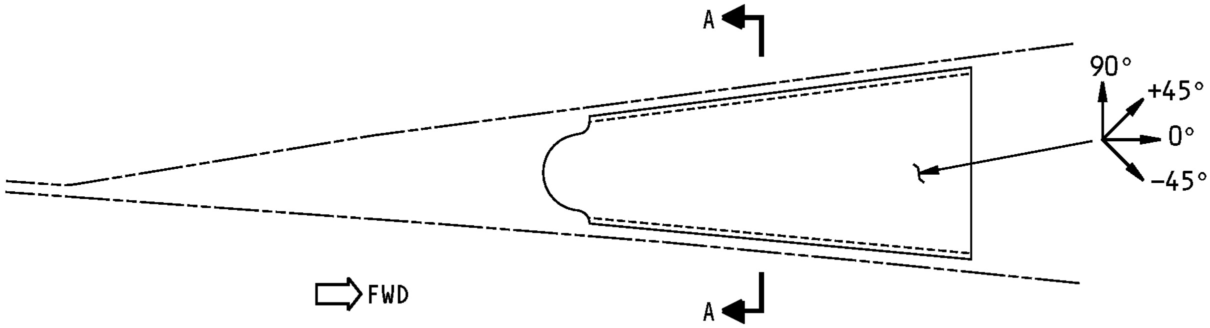
- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

**Ply Direction and Ply Sequence for the Actuator Backup Rib, Figure 2, Item [6]**
**Figure 9**

**Table 9**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 9 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P5, P8, P10, P12 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P4, P6, P7, P9, P11 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

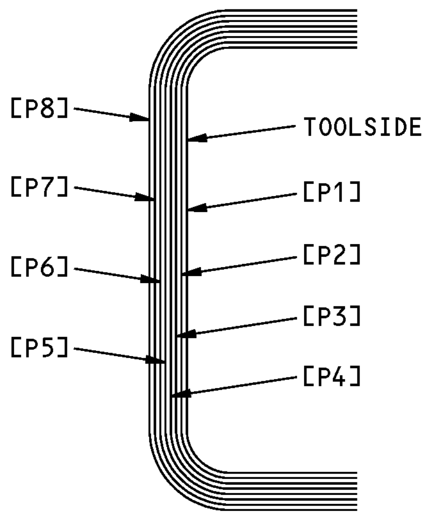



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED SUPPORT RIB

PLY LAYUP DIRECTION

A




PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

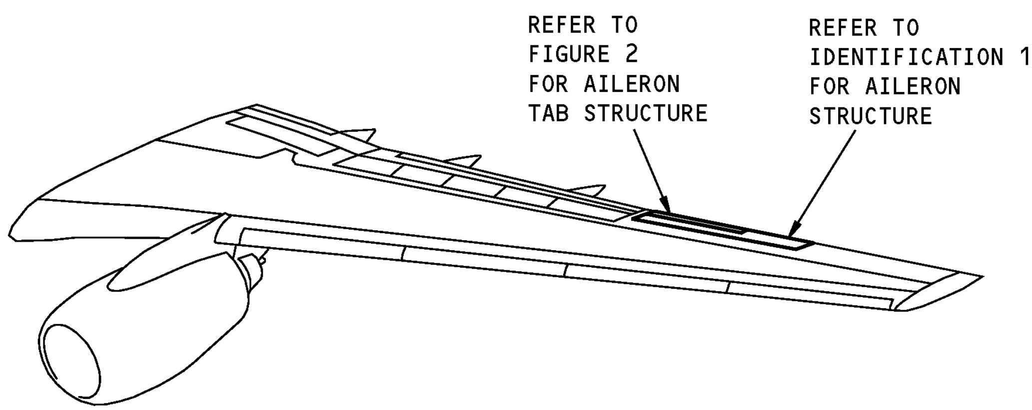
**Ply Direction and Ply Sequence for the Bonded Support Rib, Figure 2, Item [7]**
**Figure 10**

**Table 10**:

| PLY MATERIAL AND DIRECTION FOR FIGURE 10 | | |
|---|---|---|
| **PLY** | **DIRECTION** | **MATERIAL** |
| P1, P3, P6, P8 | + or - 45 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |
| P2, P4, P5, P7 | 0 or 90 degrees | CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW |

**NOTE**: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

## IDENTIFICATION 2 - AILERON TAB STRUCTURE



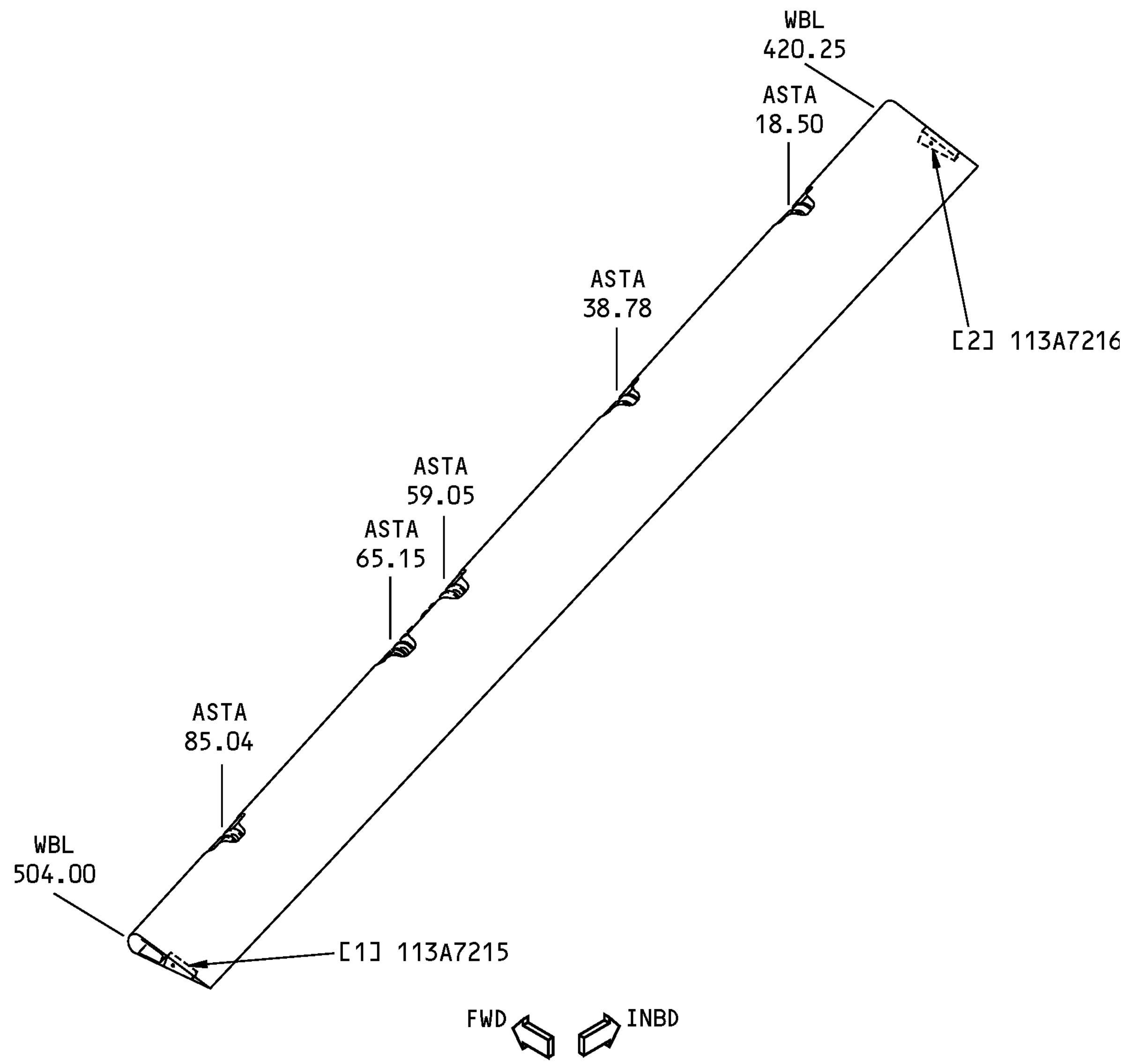

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Aileron Tab Structure Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 113A7001 | Aileron Installation |
| 113A7002 | Aileron/Tab Assembly |
| 113A7200 | Tab Assembly - Aileron |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE SHOWN, RIGHT SIDE OPPOSITE

AILERON TAB ASSEMBLY

**Aileron Tab Structure Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Outboard Closure Rib | 0.625 (15.9) | Phenolic Sheet as given in ASTM D709 (Optional: CIBA-GEIGY Resin ARALDITE and HY956 Hardener) | |
| [2] | Inboard Closure Rib | 0.625 (15.9) | Phenolic Sheet as given in ASTM D709 (Optional: CIBA-GEIGY Resin ARALDITE and HY956 Hardener) | |

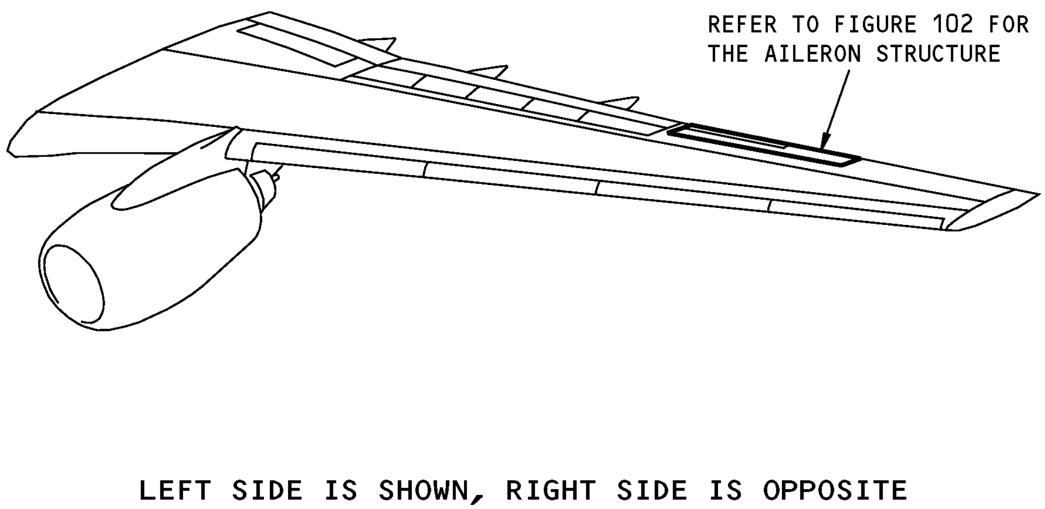
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## ALLOWABLE DAMAGE 1 - AILERON STRUCTURE

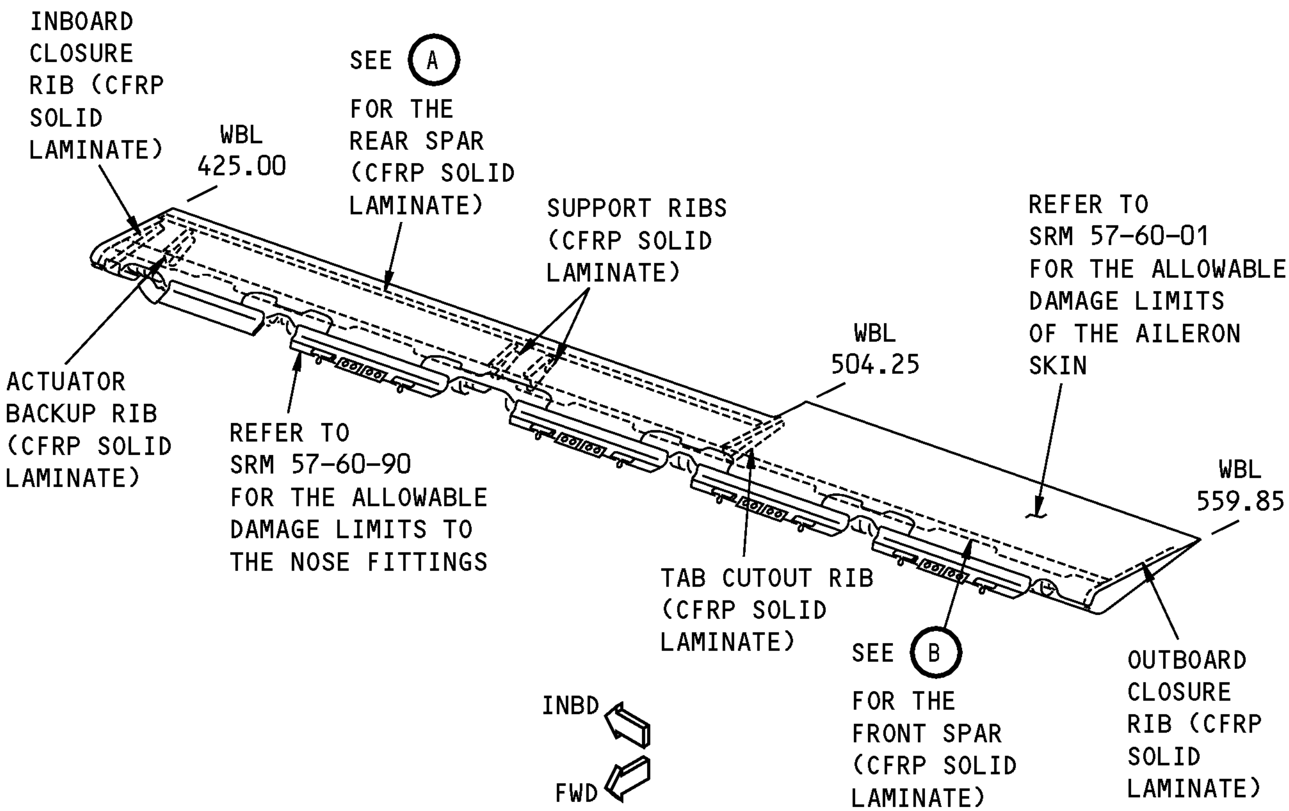
### 1. Applicability

A. This subject gives the allowable damage limits for the aileron structure shown in Wing Aileron Structure Location, Figure 101/ALLOWABLE DAMAGE 1.

B. The allowable damage limits are only applicable if they are sealed as given in Paragraph 2.

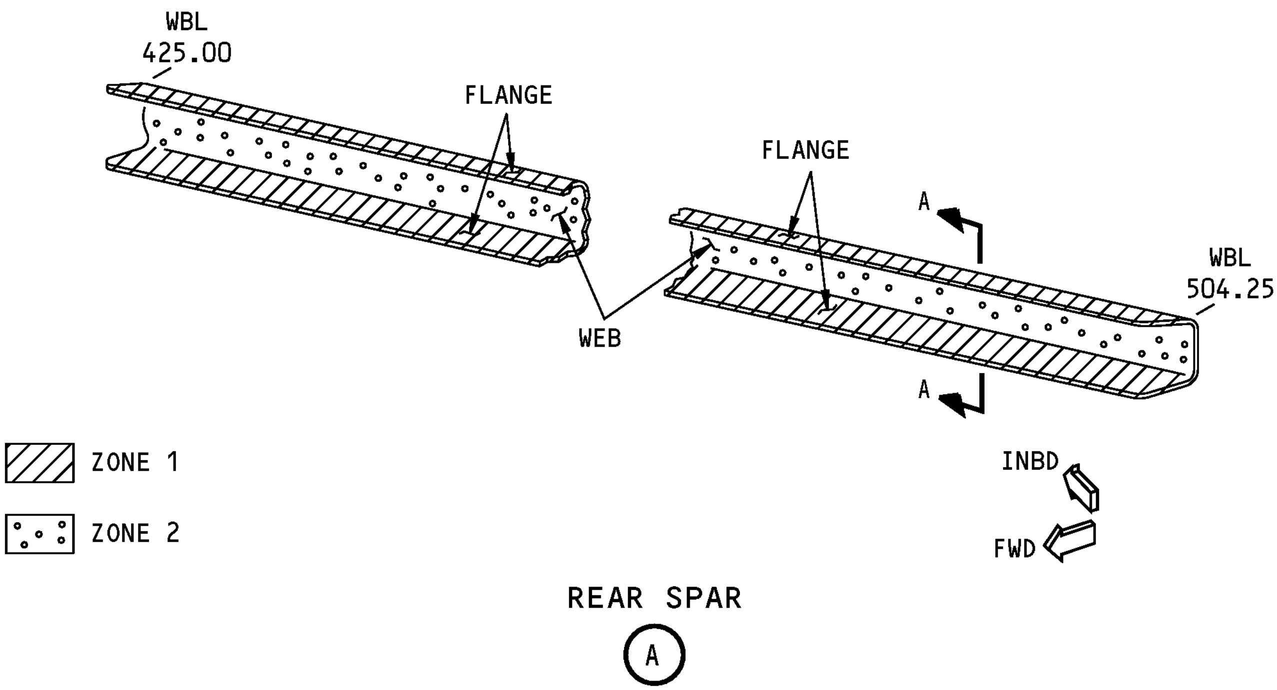


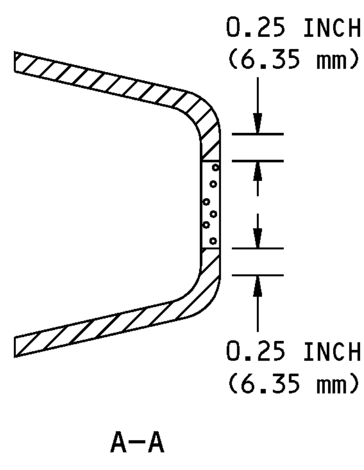

**Wing Aileron Structure Location**
**Figure 101**



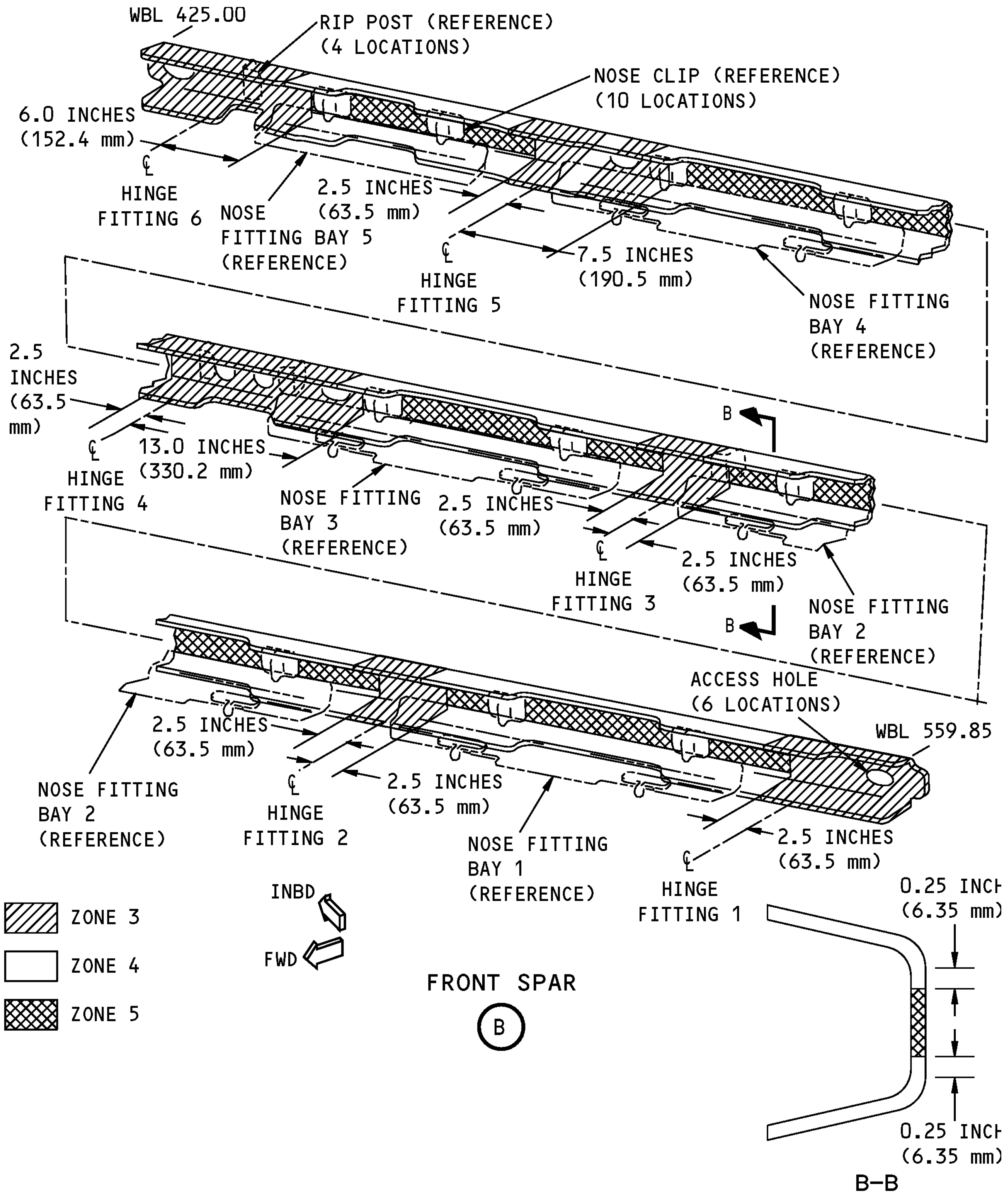

LEFT SIDE UPPER SURFACE IS SHOWN,
RIGHT SIDE IS OPPOSITE

**Aileron Structure**
**Figure 102 (Sheet 1 of 3)**




**Aileron Structure**
**Figure 102 (Sheet 2 of 3)**




**Aileron Structure**
**Figure 102 (Sheet 3 of 3)**

2. **General**

A. Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two methods that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

(d) Repair the damage at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Use a vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04.

(2) Make a temporary seal with aluminum foil tape (speed tape).

(3) Keep a record of the location.

(4) Repair the damage at or before 400 flight hours from the time the seal was made.

E. The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause carbon fiber damage and are sealed.

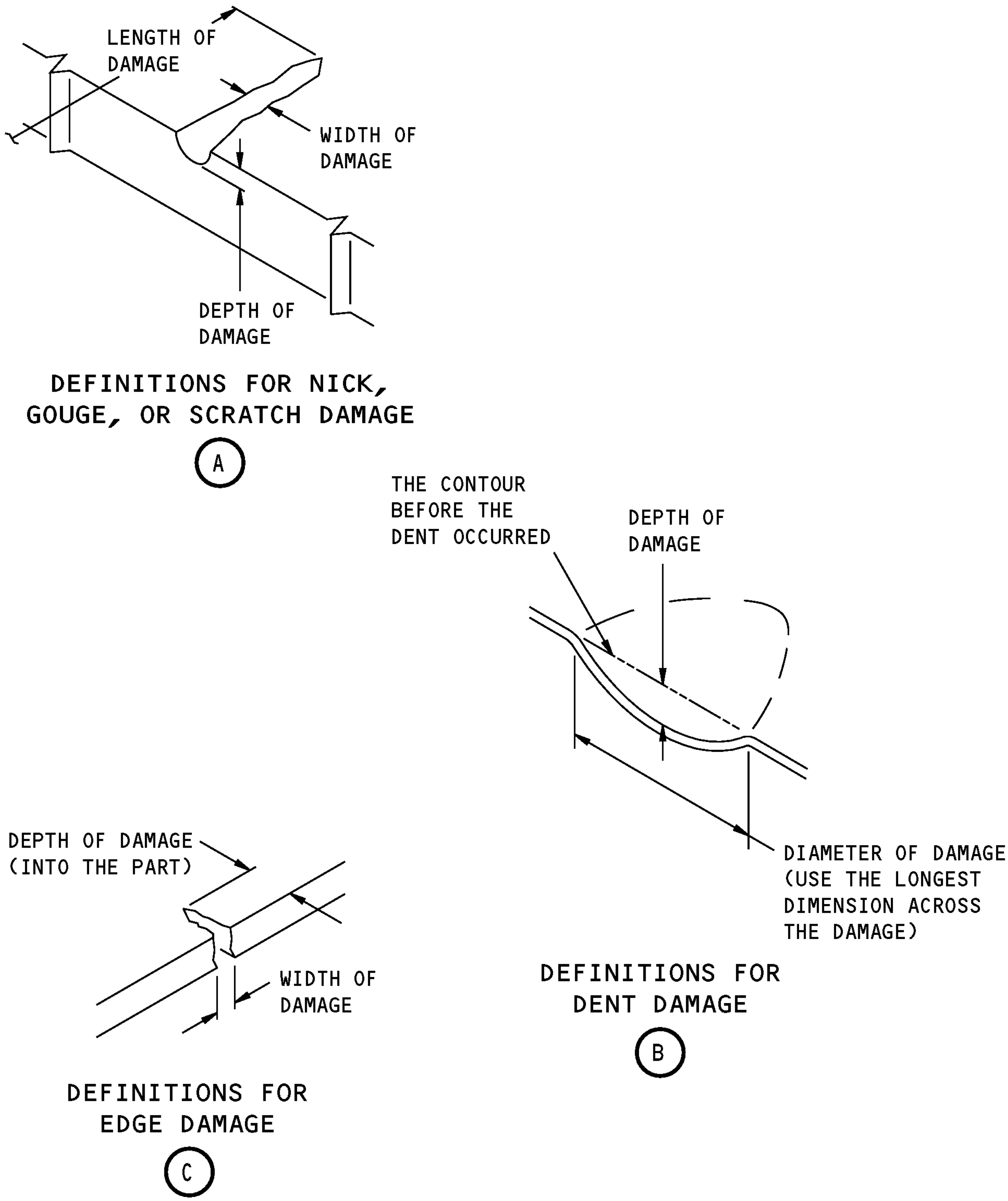
F. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

G. Refer to Table 101/ALLOWABLE DAMAGE 1 for the references for the allowable damage limits of the different parts.

**Table 101**:

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| **TYPE OF STRUCTURE** | **ZONE LOCATION** | **PARAGRAPH** |
| Ribs | — | 4.A |

| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| TYPE OF STRUCTURE | ZONE LOCATION | PARAGRAPH |
| Rear Spar | 1 | 4.B |
| | 2 | 4.C |
| Front Spar | 3 | 4.D |
| | 4 | 4.E |
| | 5 | 4.F |




**Damage Definitions**
**Figure 103**

## 3. References

| Reference | Title |
| --- | --- |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-60-01 | AILERON BALANCING PROCEDURES |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-08 | RESIN SWEEP-FAIR PROCEDURES |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

## 4. Allowable Damage Limits

A. Ribs

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the chord radius and are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.5 inch (12.7 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(4) Dents are permitted if they are not in the chord radius and are:

(a) A maximum of 0.05 inch (1.27 mm) in depth

(b) A maximum of 2.0 inch (50.8 mm) in diameter

(c) A minimum of 3.00 inches (76.2 mm) away from other damage

(d) A minimum of 3.00 inches (76.2 mm) away from a rib post.

(5) Holes and Punctures are permitted that are not in the chord radius if they are:

(a) A maximum of 0.25 inch (6.4 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(6) Delamination is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1 if it is:

(a) Not in the chord radius

(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(7) Edge damage is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1 if it is:

(a) A maximum of 0.5 inch (12.7 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

B. Rear Spar - Zone 1

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the radius and they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.5 inch (12.7 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are not in the radius and are:

1) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter

3) A minimum of 3.00 inches (76.2 mm) away from other damage

4) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(5) Holes and Punctures are permitted if they are not in the radius and are:

(a) A maximum of 0.25 inch (6.4 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(6) Delamination is permitted as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) Not in the radius

(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(7) Edge damage is permitted as shown as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 0.5 inch (12.7 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width.

(8) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail B .

C. Rear Spar - Zone 2

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch (6.35 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter

3) A minimum of 3.00 inches (76.2 mm) away from other damage

4) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(5) Holes and Punctures are permitted if they are:

(a) A maximum of 0.1 inch (2.54 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(6) Delamination is permitted as shown in Figure 104/ALLOWABLE DAMAGE 1, Detail A .

D. Front Spar - Zone 3

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the radius and are:

(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch (6.35 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are not in the radius and are:

1) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter
3) A minimum of 3.00 inches (76.2 mm) away from other damage
4) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(5) Holes and Punctures are permitted if they are not in the radius if and are:
(a) A maximum of 0.1 inch (2.54 mm) in diameter
(b) A minimum of 3.00 inches (76.2 mm) away from other damage
(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(6) Delamination is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:
(a) Not in the radius
(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(7) Edge damage is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:
(a) A maximum of 0.25 inch (6.35 mm) in width
(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(8) Edge Erosion is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail B .

E. Front Spar - Zone 4

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the radius and are:
(a) A maximum of one ply in depth

**NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch (6.35 mm) in length
(c) A maximum of 0.25 inch (6.35 mm) in width
(d) A minimum of 3.00 inches (76.2 mm) away from other damage
(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:
(a) Dents that do not cause damage to the carbon fibers are permitted if they are not in the radius and are:
1) A maximum of 0.05 inch (1.27 mm) in depth

**NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter
3) A minimum of 3.00 inches (76.2 mm) away from other damage
4) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(5) Holes and Punctures are permitted if they are not in the radius and are:
(a) A maximum of 0.1 inch (2.54 mm) in diameter
(b) A minimum of 3.00 inches (76.2 mm) away from other damage
(c) A minimum of 3.00 inches (76.2 mm) away from a fitting.

(6) Delamination is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:
  (a) Not in the radius
  (b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.
(7) Edge damage is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:
  (a) A maximum of 0.25 inch (6.35 mm) in depth
  (b) A maximum of 0.25 inch (6.35 mm) in width
  (c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.
(8) Edge Erosion is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail B .

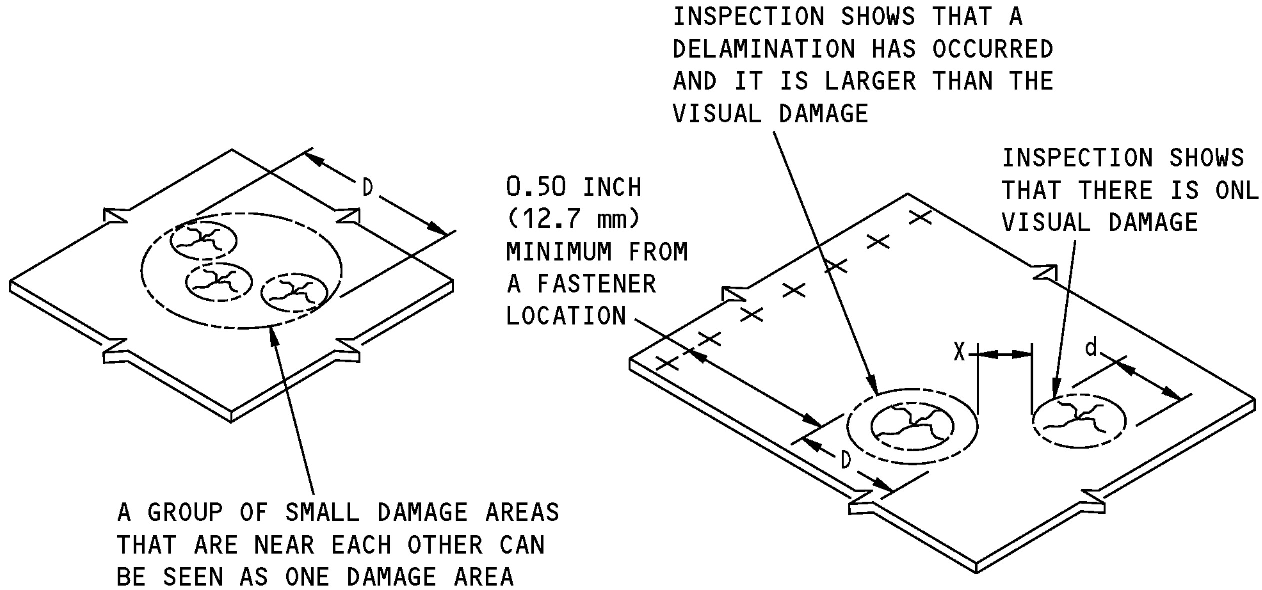
F. Front Spar - Zone 5
(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.
(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.
(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:
  (a) A maximum of one ply in depth

  **NOTE**: Use the limits for holes and punctures if the damage is more than one ply in depth.

  (b) A maximum of 0.25 inch (6.35 mm) in length
  (c) A maximum of 0.25 inch (6.35 mm) in width
  (d) A minimum of 3.00 inches (76.2 mm) away from other damage
  (e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.
(4) Dents:
  (a) Dents that do not cause damage to the carbon fibers are permitted if they are:
    1) A maximum of 0.05 inch (1.27 mm) in depth

    **NOTE**: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

    2) A maximum of 0.50 inch (12.7 mm) in diameter
    3) A minimum of 3.00 inches (76.2 mm) away from other damage
    4) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.
(5) Holes and Punctures are permitted if they are:
  (a) A maximum of 0.1 inch (2.54 mm) in diameter
  (b) A minimum of 3.00 inches (76.2 mm) away from other damage
  (c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.
(6) Delamination is permitted as shown in Figure 104/ALLOWABLE DAMAGE 1, Detail A .




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

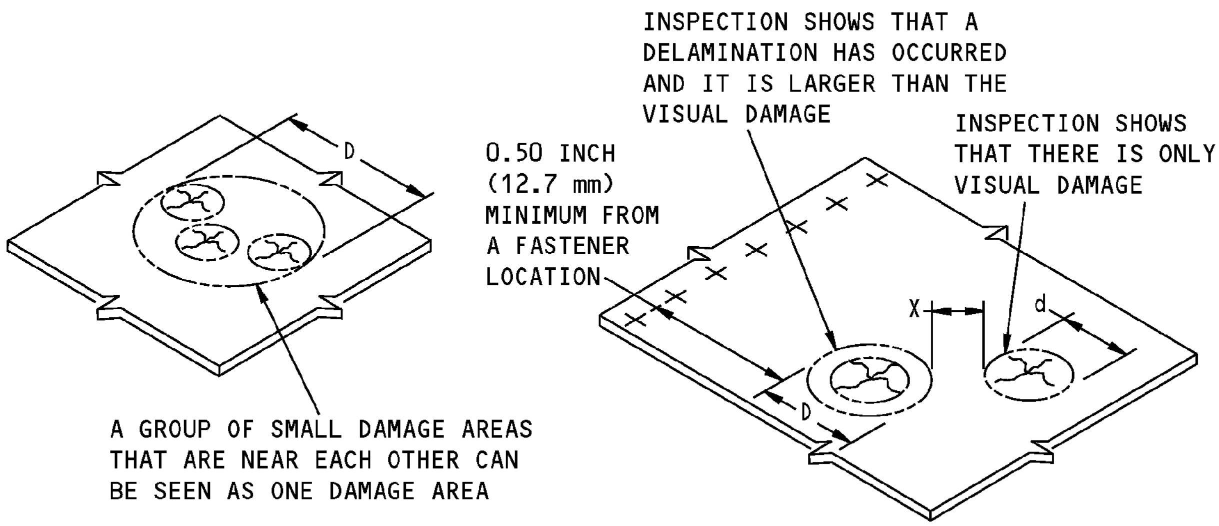
d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.00 INCHES (76.2 mm).

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

**Allowable Damage Limits**
**Figure 104**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

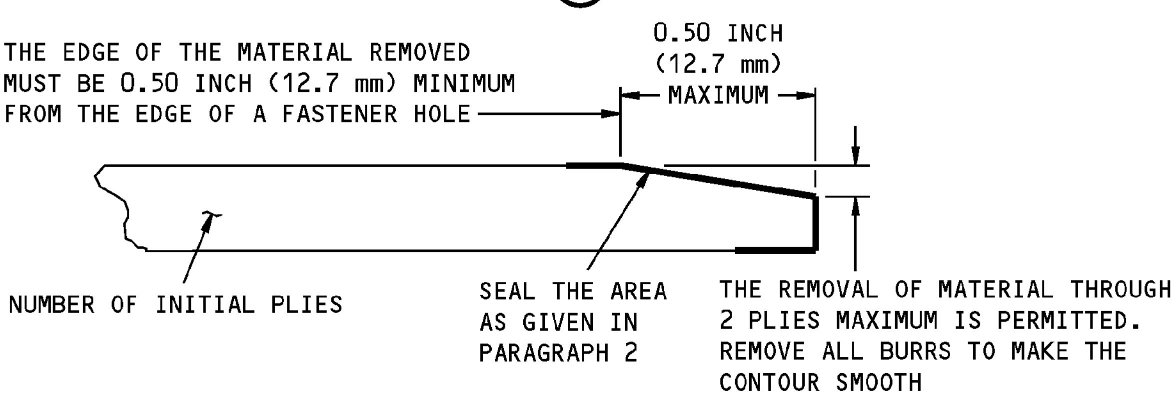
d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.00 INCHES (76.2 mm).

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A




CLEANUP AND SEALING OF EDGE EROSION

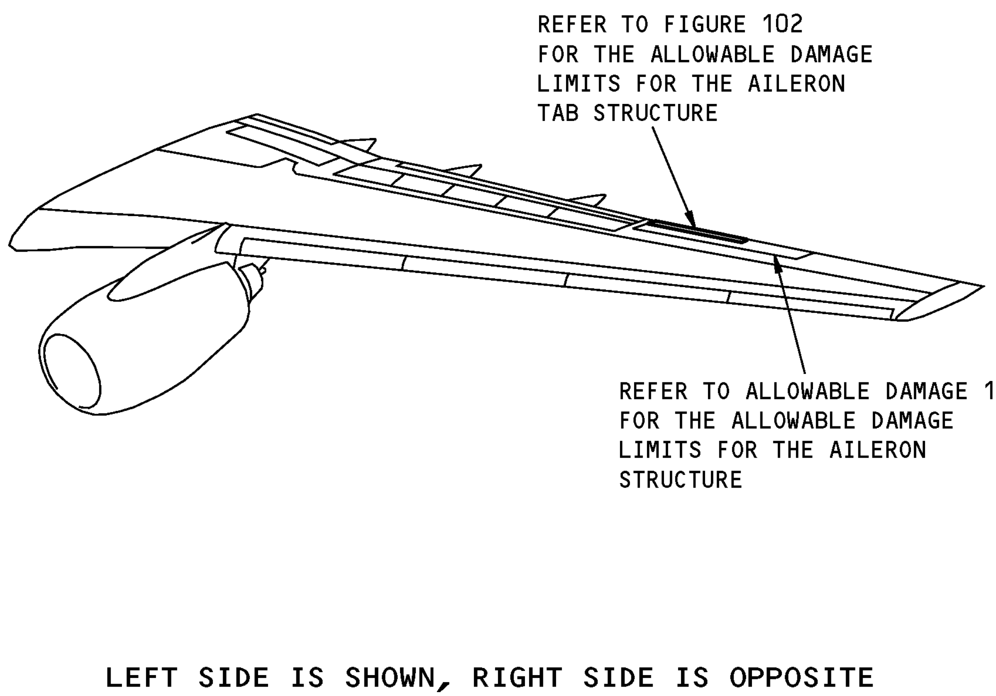
B

**Allowable Damage Limits**
**Figure 105**

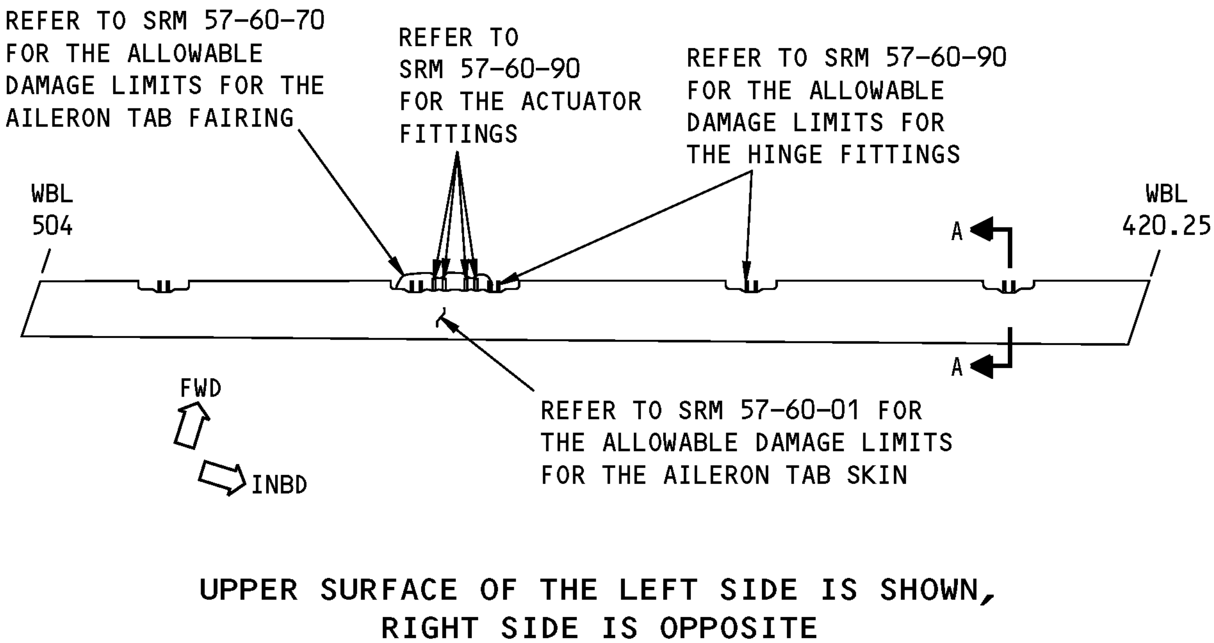
## ALLOWABLE DAMAGE 2 - AILERON TAB STRUCTURE

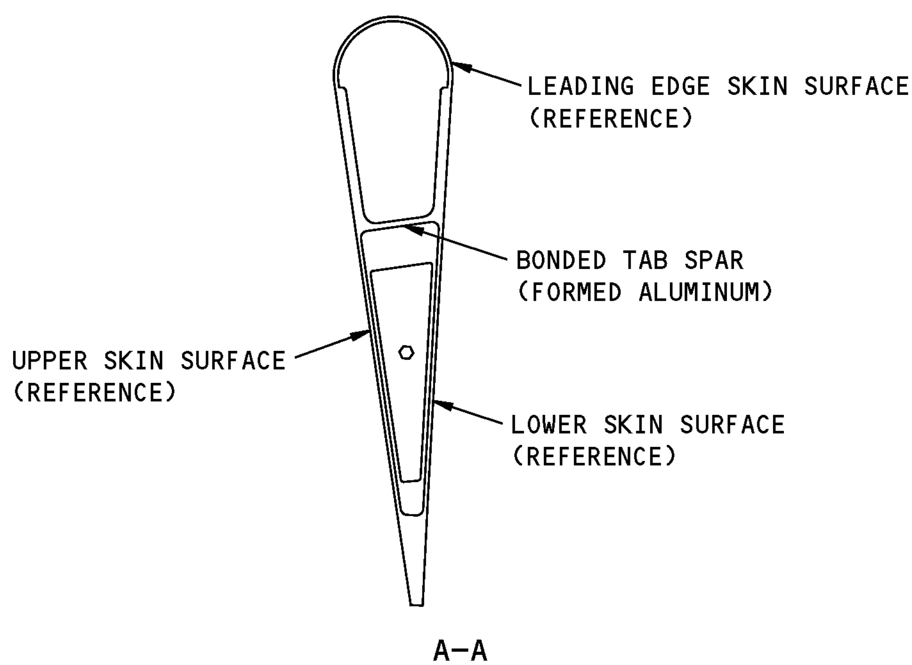
### 1. Applicability

A. This subject gives the allowable damage limits for the aileron tab structure shown in Aileron Tab Structure Location, Figure 101/ALLOWABLE DAMAGE 2.




**Aileron Tab Structure Location**
**Figure 101**




**Aileron Tab Structure**
**Figure 102**

**2. General**

A. Do the steps that follow.

(1) Removed the damaged material.

(a) Refer to 51-10-02 for the procedures.

(b) Refer to 51-30-03 for the sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for the sources of equipment and tools you can use to remove the damage.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply one layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

**3. References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

**4. Allowable Damage Limits**

A. Nose Spar

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail F .

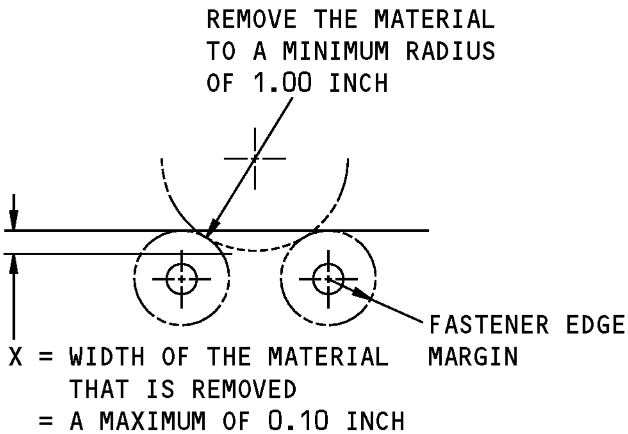
(4) Holes and Punctures:

(a) Damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

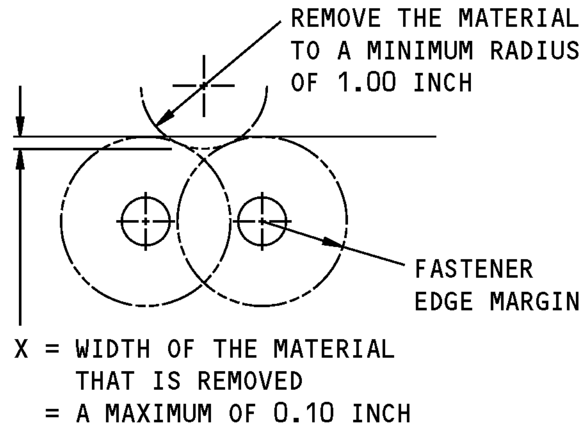
2) A minimum of 4D (D = The diameter of the hole or puncture) from a hole, edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet installed without sealant.



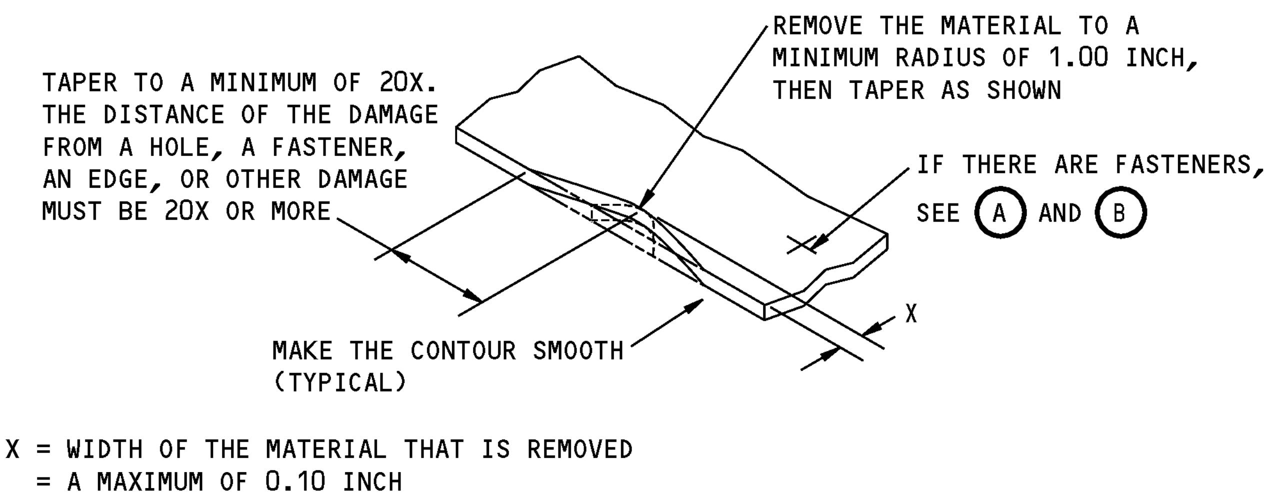

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

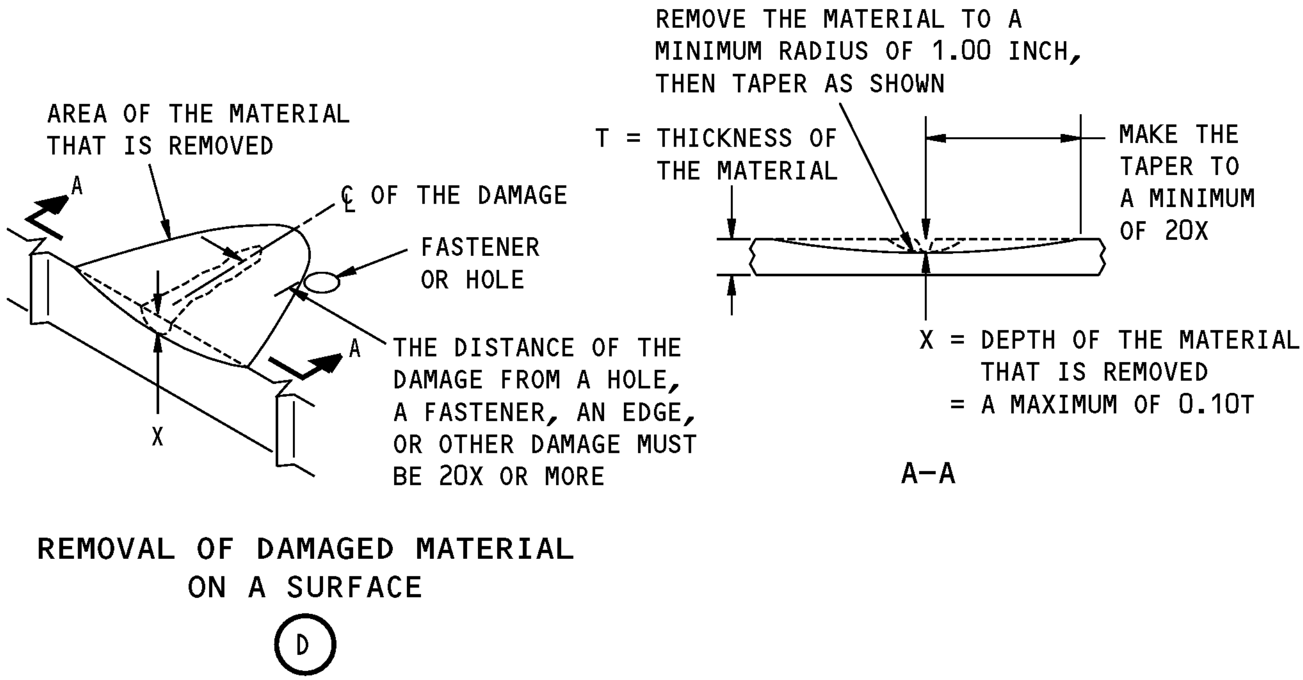
B

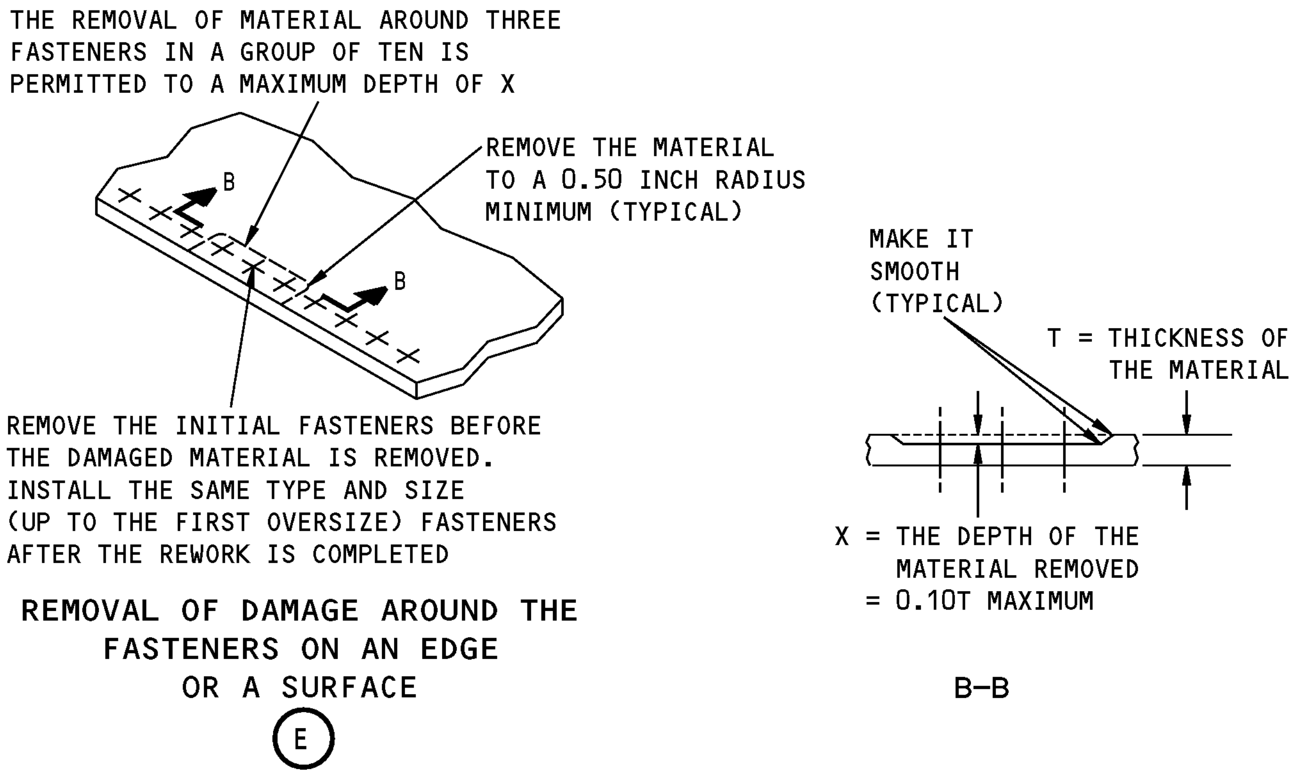



REMOVAL OF DAMAGED MATERIAL ON AN EDGE

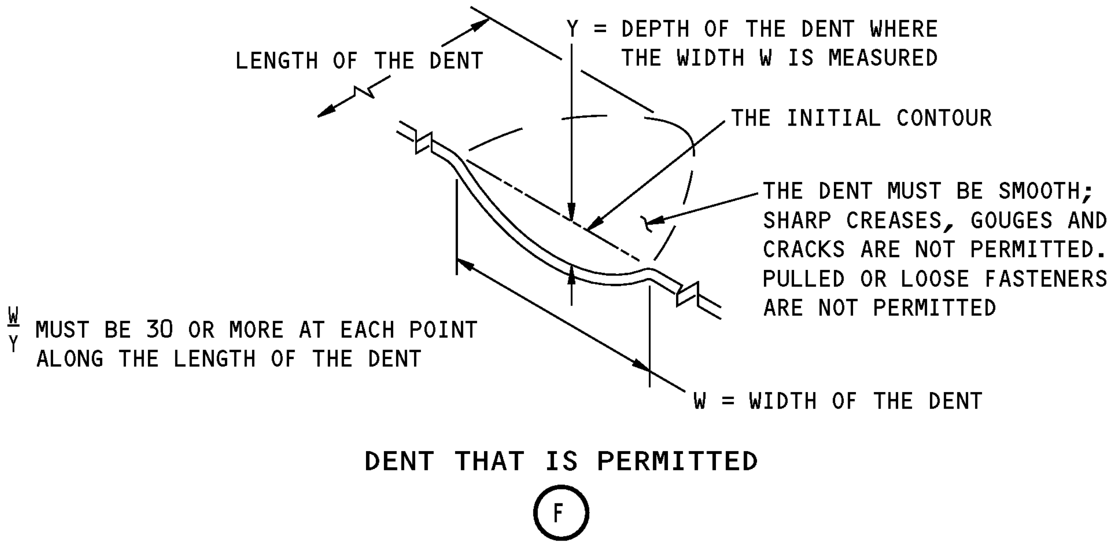
E

**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 3)**




**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 3)**




**Allowable Damage Limits**
**Figure 103 (Sheet 3 of 3)**

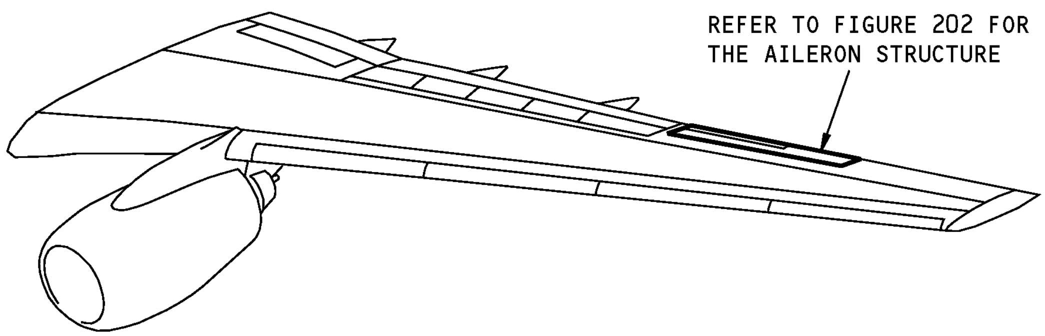
## REPAIR 1 - AILERON STRUCTURE

### 1. Applicability

A. Repair 1 is applicable to the front and rear spars of the aileron shown in Wing Aileron Structure Repair, Figure 201/REPAIR 1.

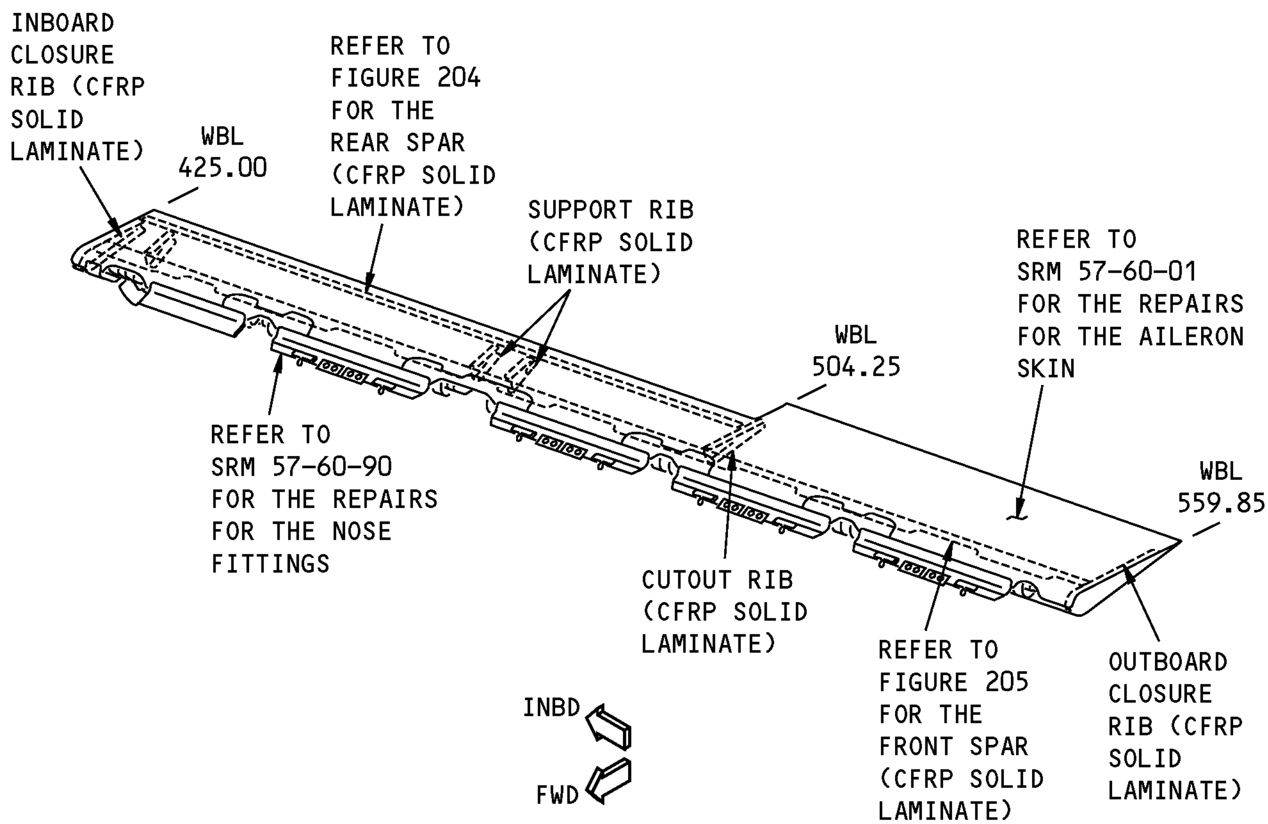
(1) Boeing has not found it necessary to supply repairs to the ribs of the aileron in the Structural Repair Manual (SRM) at this time.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Aileron Structure Repair**
**Figure 201**




LEFT SIDE UPPER SURFACE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Aileron Structure Repair**
**Figure 202**

2. **General**

A. Repair 1 gives instructions for Category B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the aileron as necessary. Refer to AMM 27-11-11/401.

C. Remove the skin as necessary to get access to the spar.

(1) If a fastener is damaged, refer to 51-70-04 or 51-70-05 for a repair.

D. Remove the fitting as necessary to get access to the damaged area. Refer to SOPM 20-10-08.

E. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to NDT, Part 1, 51-01-02 for inspection procedures.

**NOTE**: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 203/REPAIR 1 for the definitions of the diameter and depth of damage.

F. Do the repair as given in Paragraph 4./REPAIR 1

G. Install the fittings if they were removed.

(1) Apply BMS 5-95 sealant to:

(a) The reworked areas of the spar and fittings

(b) The edges around the fittings

(c) All parts that connect to the spar and fittings

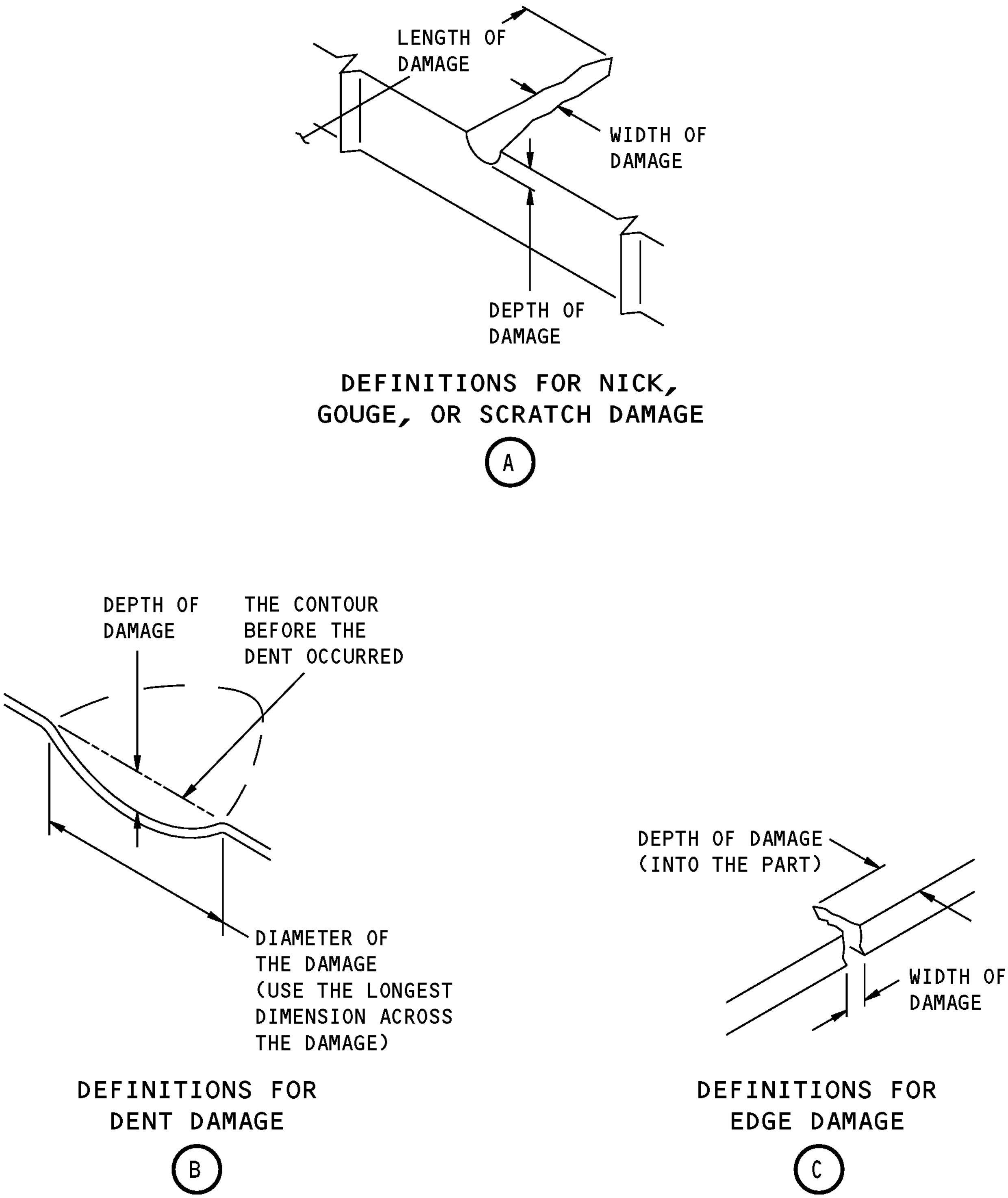
(d) All gaps.

(2) Apply BMS 10-79, Type III primer to the edges around the fittings. Refer to SOPM 20-44-04.

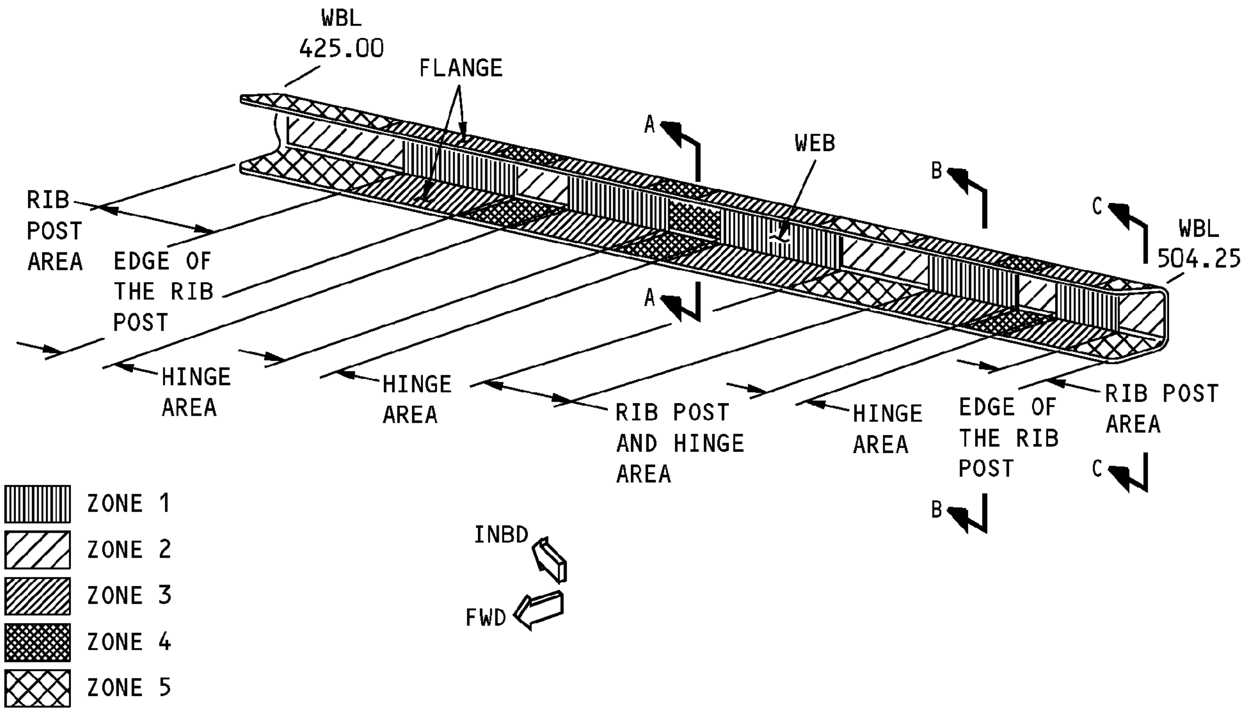
(3) Apply BMS 10-60, Type II primer to the edges around the fittings. Refer to AMM PAGEBLOCK 51-21-99/701.

H. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

I. Install the aileron, as applicable. Refer to AMM 27-11-11/401.



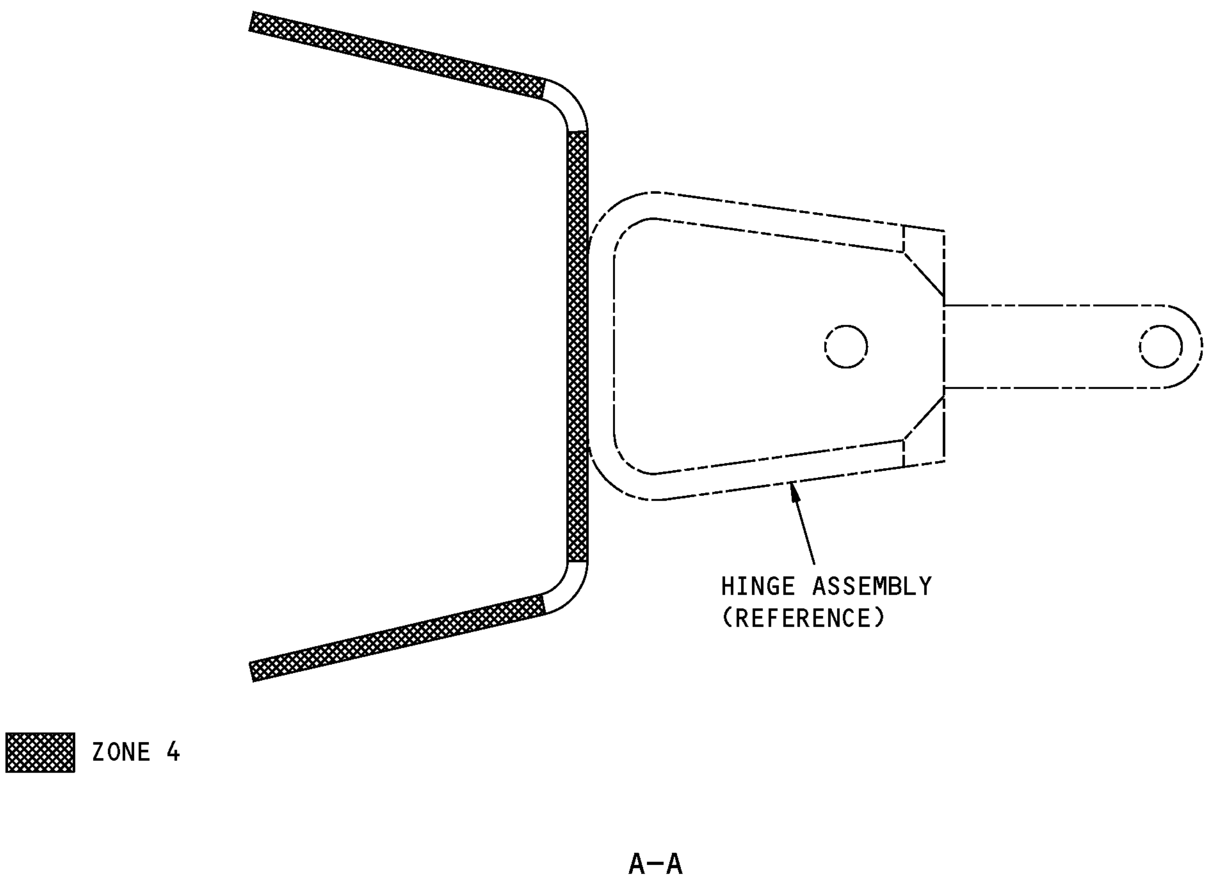

**Damage Definitions**
**Figure 203**



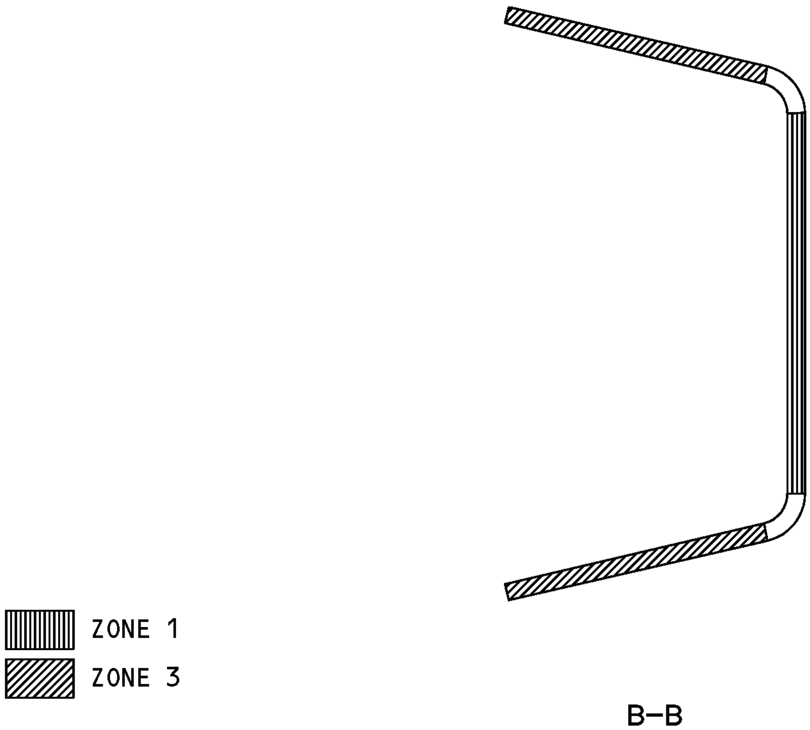

NOTE: THE EDGE OF THE HINGE AREAS ALIGN WITH THE EDGE OF THE HINGE FITTINGS.

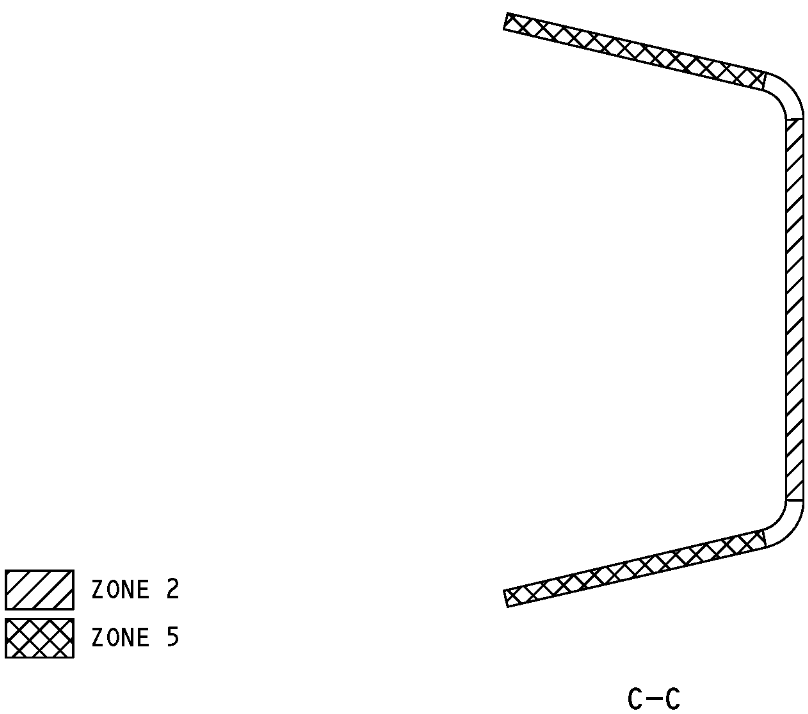
REAR SPAR

**Aileron Structure Repair Zones - Rear Spar**
**Figure 204 (Sheet 1 of 3)**

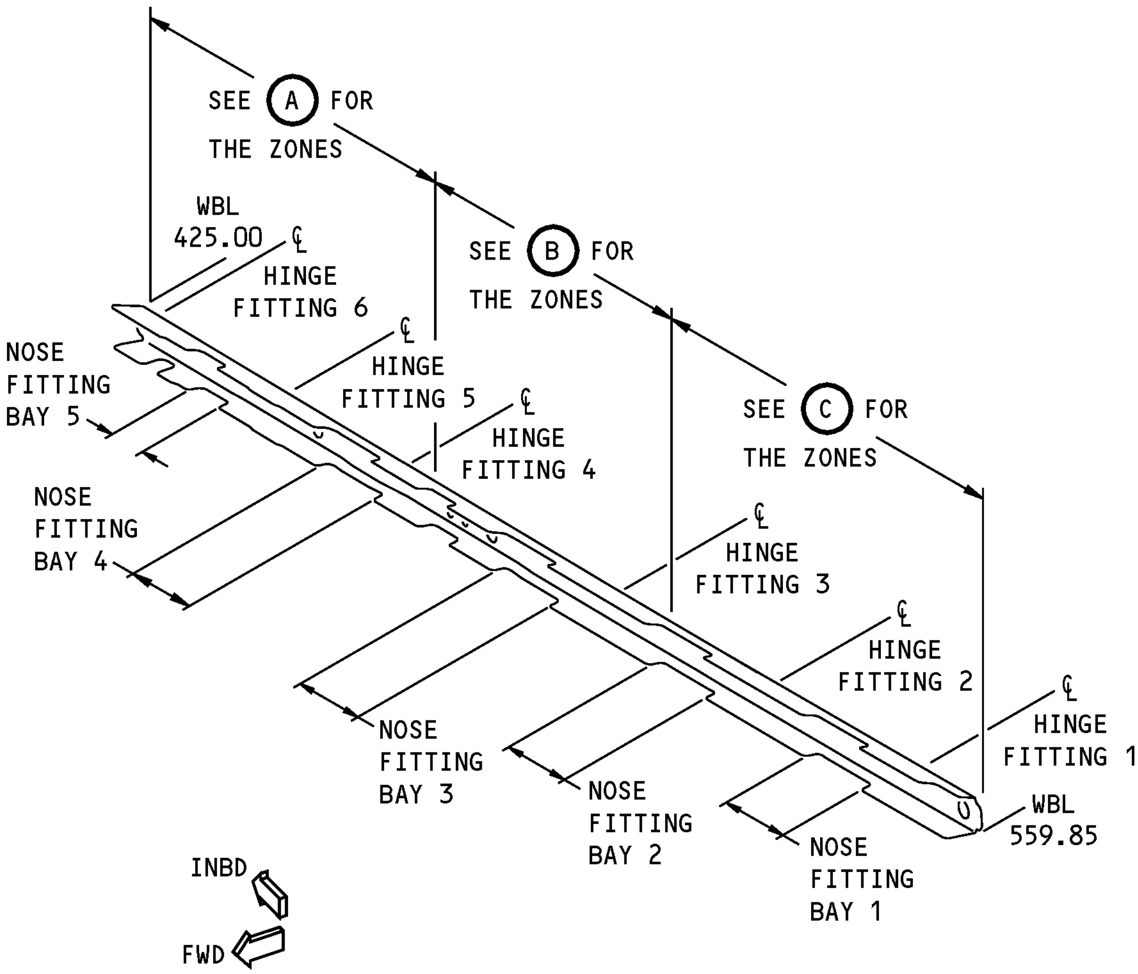



**Aileron Structure Repair Zones - Rear Spar**
**Figure 204 (Sheet 2 of 3)**

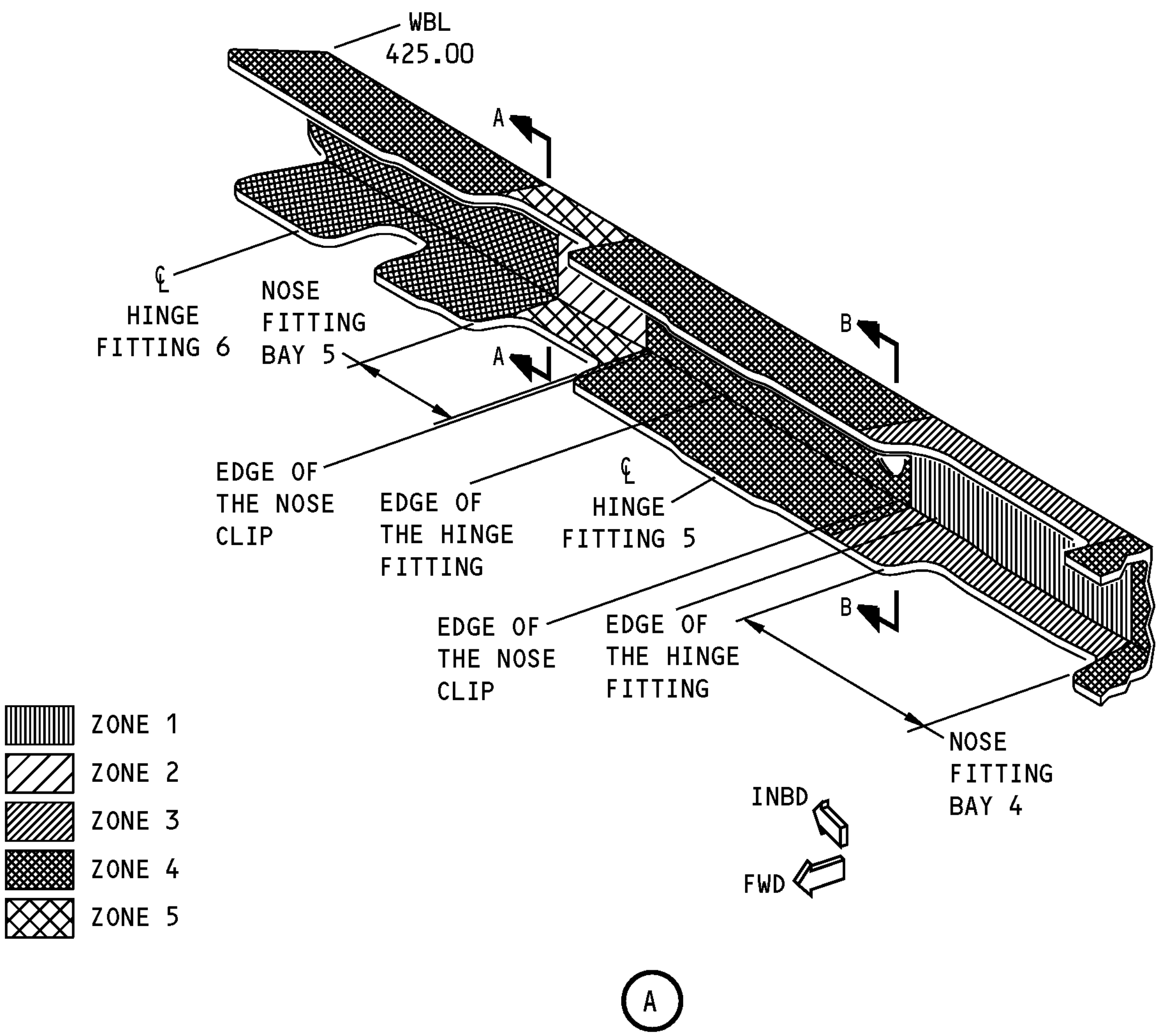


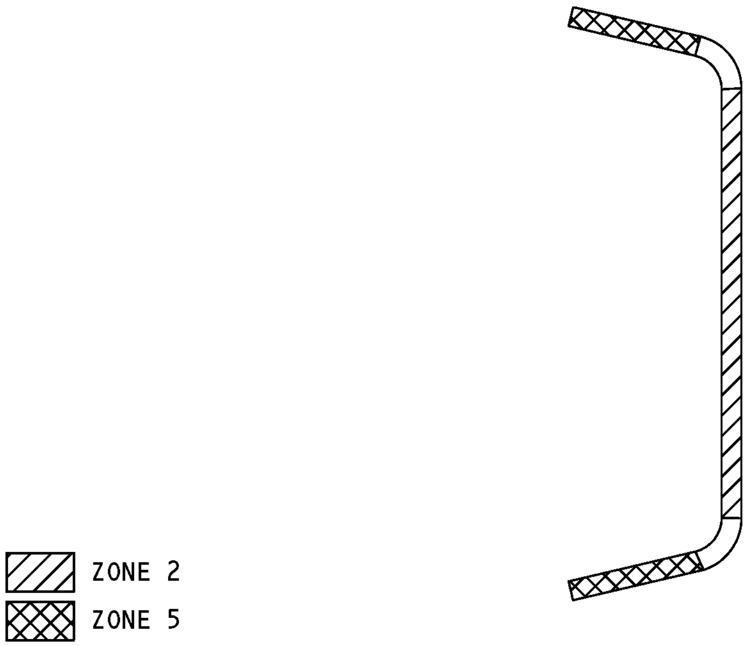

**Aileron Structure Repair Zones - Rear Spar**
**Figure 204 (Sheet 3 of 3)**



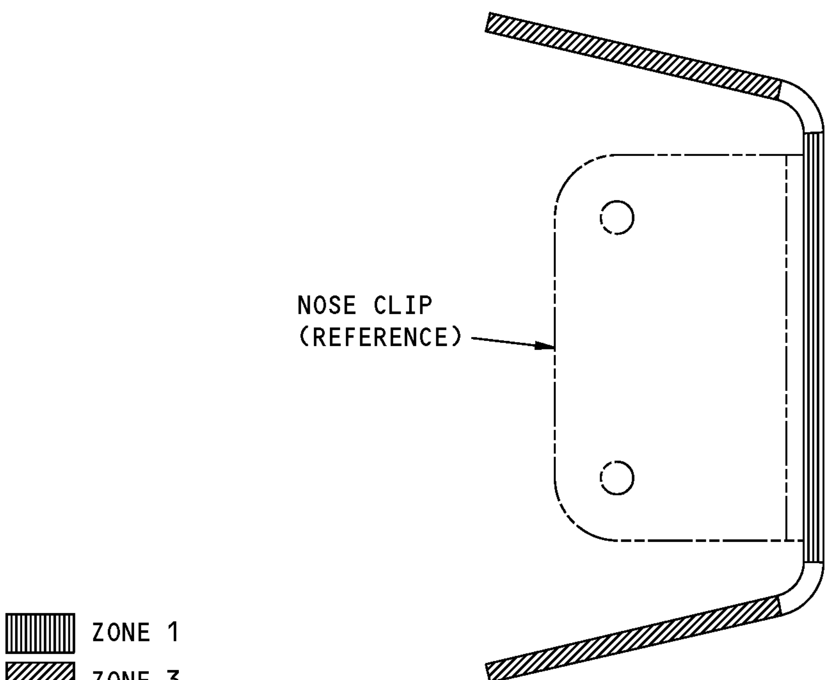

**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 1 of 8)**




**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 2 of 8)**

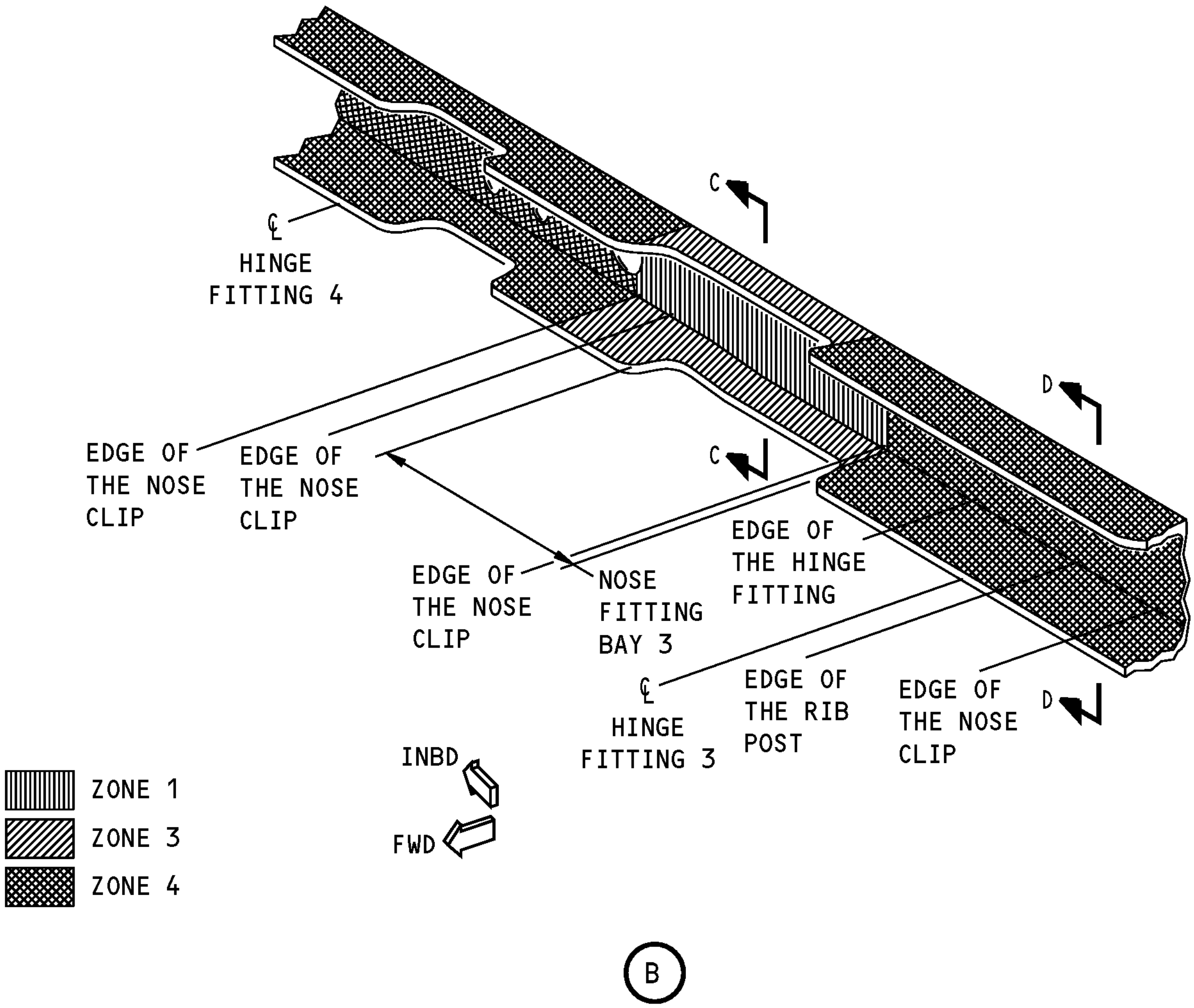



A-A

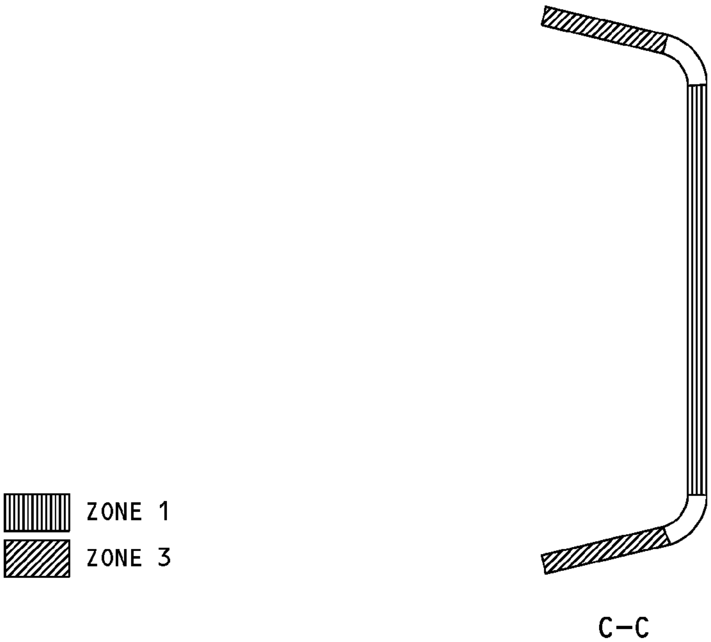


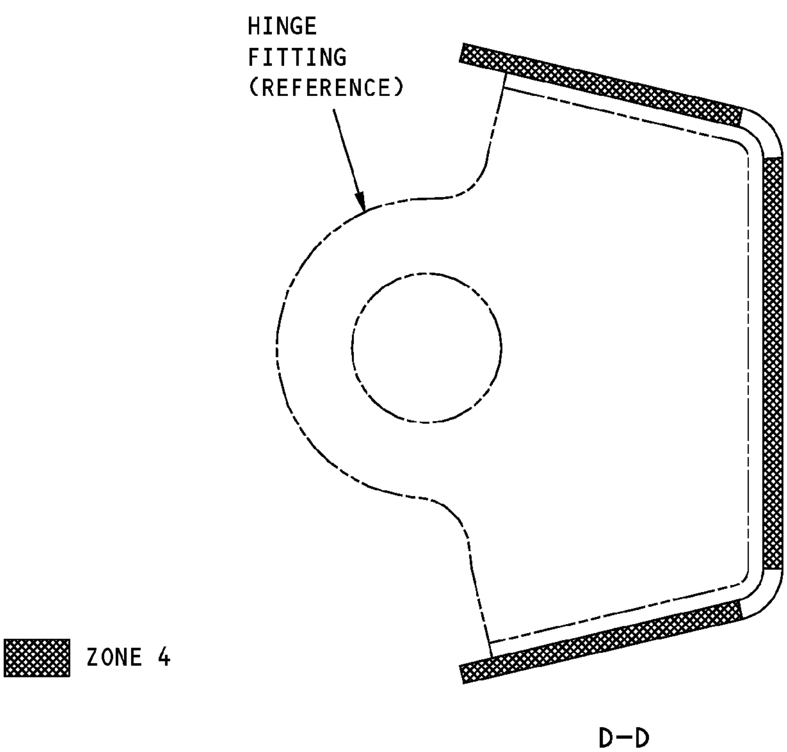

B-B

**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 3 of 8)**

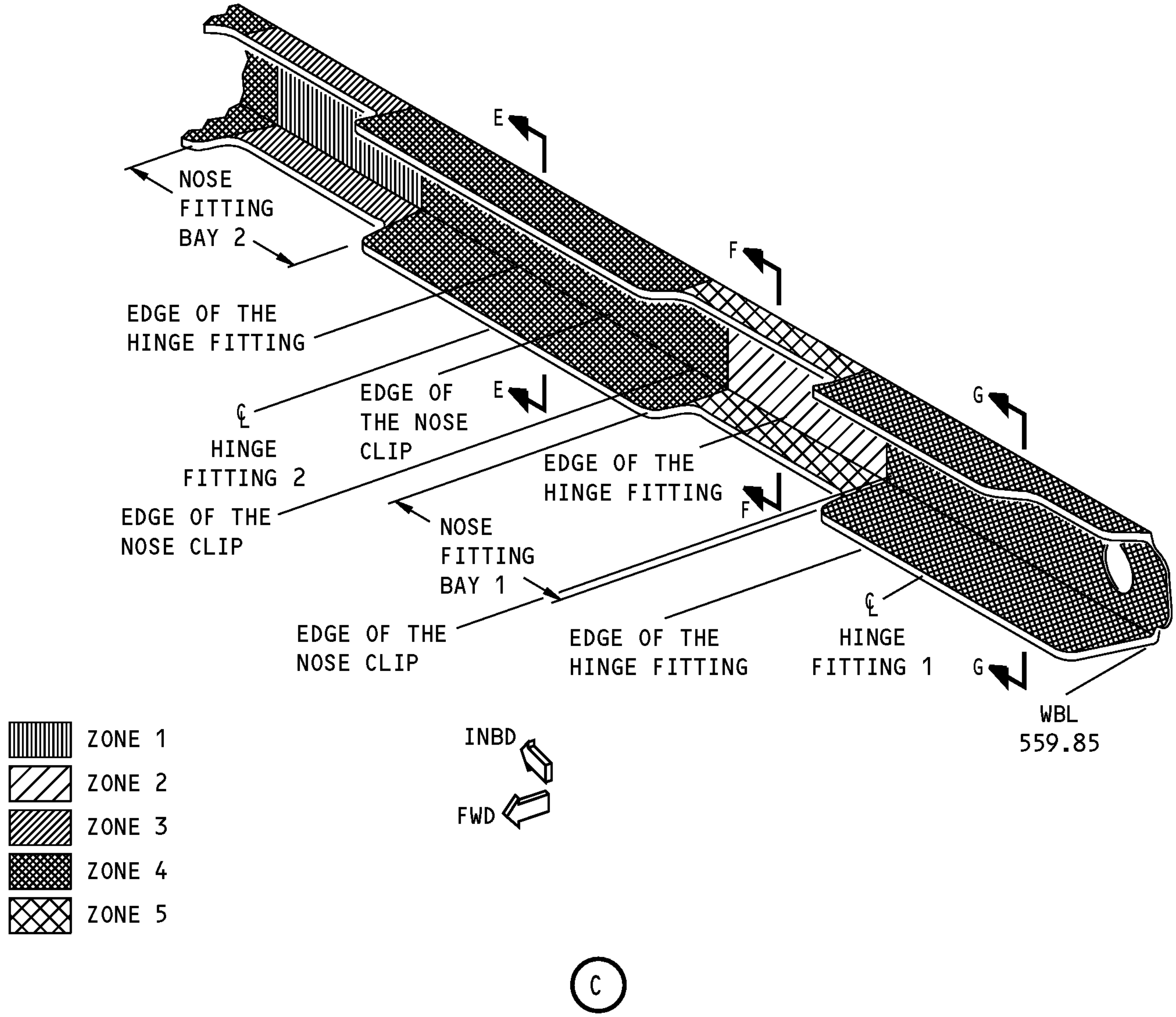



**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 4 of 8)**

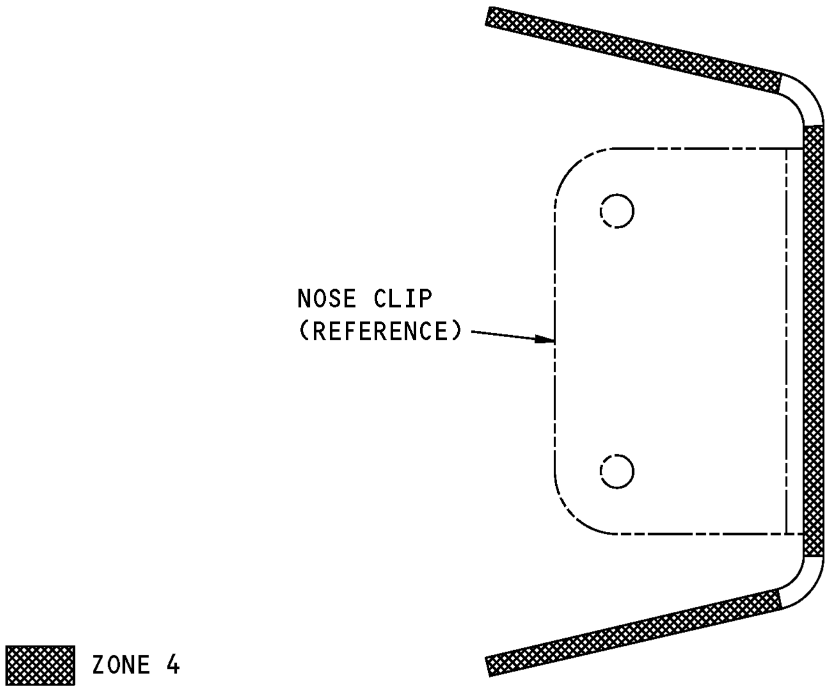


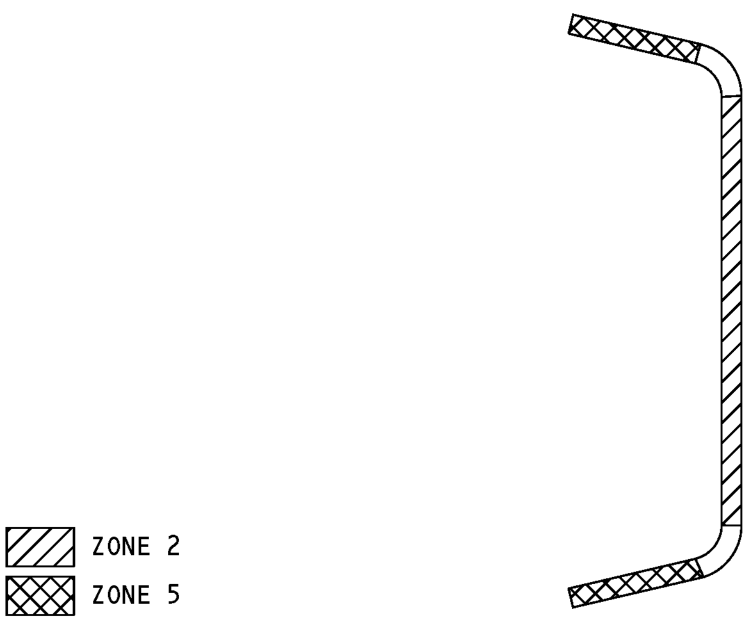

**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 5 of 8)**




**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 6 of 8)**

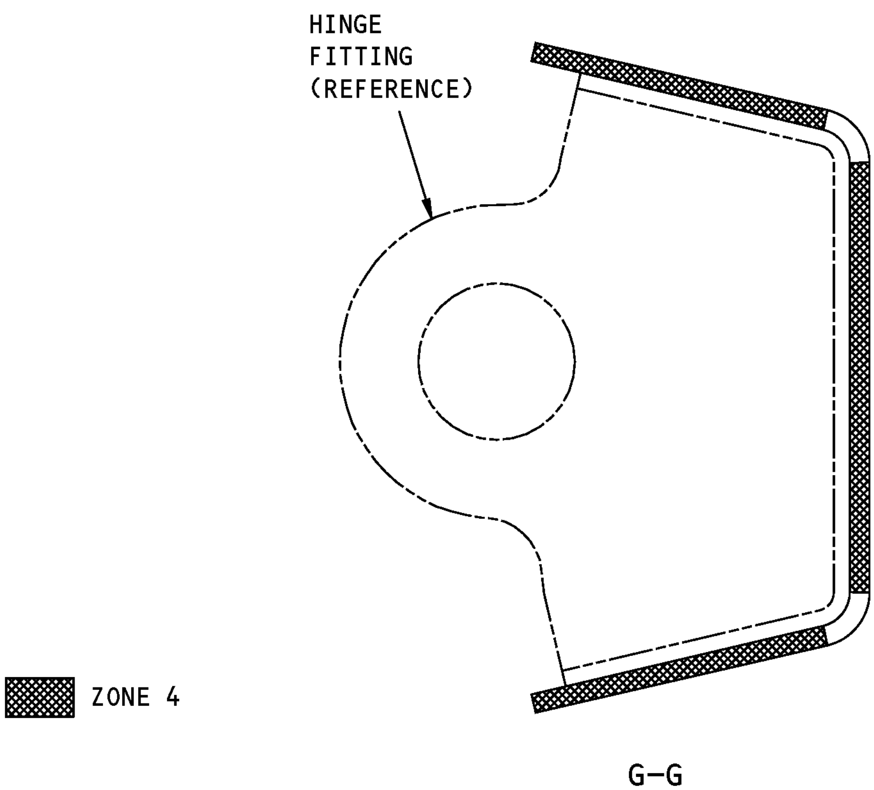



E–E




F–F

**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 7 of 8)**




**Aileron Structure Repair Zones - Front Spar**
**Figure 205 (Sheet 8 of 8)**

## 3. References

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-01 | AERODYNAMIC SMOOTHNESS |
| 51-20-05 | REPAIR SEALING |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-60-01 | AILERON BALANCING PROCEDURES |
| 51-70-04 | REPAIR PROCEDURES FOR WET LAYUP MATERIALS |
| 51-70-05 | REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| AMM 27-11-11/401 | Aileron - Removal/Installation |
| SOPM 20-10-08 | Removal of Faying Surface Sealed Metal Fittings from Composite Structures |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 1, 51-01-02 | NDT Examination of Composite Structure for Impact Damage |

## 4. Repair Instructions

A. If a dent is 2.00 inches in diameter or less, and has no fiber damage or delamination, then do as follows:

(1) Fill the dent with potting compound.

(2) Apply a fiberglass patch as given in Repair 14 of 51-70-04.

B. Refer to the Tables that follow for all other damage:

(1) Table 201/REPAIR 1 for the repair data that is applicable to damage to Zone 1 of the Front and Rear Spars

(2) Table 202/REPAIR 1 for the repair data that is applicable to damage to Zone 2 of the Front and Rear Spars

(3) Table 203/REPAIR 1 for the repair data that is applicable to damage to Zone 3 of the Front and Rear Spars

(4) Table 204/REPAIR 1 for the repair data that is applicable to damage to Zone 4 of the Front and Rear Spars

(5) Table 205/REPAIR 1 for the repair data that is applicable to damage to Zone 5 of the Front and Rear Spars.

C. Do as follows when you make a repair:

(1) When you remove the damage, do not cut or make an abrasion into the radius of the structure.

(2) If the repair plies make an overlap of a hole or cutout, do the steps that follow:

(a) Cure the repair.

(b) Drill or cut the plies to the initial diameter of the hole or cutout.

(3) If you need clearance with adjacent structure, install the tapered shim on each side of the repair.

(4) It is permitted to put the repair plies around the full width of the structure.

(a) Do not make an overlap of the edges of the structure.

D. For repairs made with wet layup materials that are cured at 200° (93°C), do as follows:

(1) Use one repair ply of fabric for each initial ply that was removed from the damaged area.

(2) Add three structural plies of fabric. Put one structural ply at ±45 degrees and the other at 0 or 90 degrees.

**NOTE**: Repair plies or added plies are not necessary for edge delamination that is more than 2D (D= fastener diameter) away from a fastener hole.

(3) Add a ply of fiberglass.

E. For repairs made with preimpregnated layup materials that are cured at 250°F (121°C), do as follows:

(1) Use one repair ply of fabric for each initial ply that was removed from the damaged area.

(2) Add three structural plies of fabric. Put one structural ply at ±45 degrees and the other at 0 or 90 degrees.

**NOTE**: Repair plies or added plies are not necessary for edge delamination that is more than 2D (D= fastener diameter) away from a fastener hole.

(3) Add a ply of fiberglass.

F. For repairs made with preimpregnated layup materials that are cured at 350° (177°C), do as follows:

(1) Use one repair ply of fabric for each initial ply that was removed from the damaged area.

(2) Add a ply of fiberglass.

(3) If the damage applies to more than one zone then:

(a) The repair must not be more than the limits for each zone

(b) Do the repair instructions that are applicable to all zones.

**Table 201**:

| REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 1 | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | A repair is not permitted | Damage that is a maximum of:<br>- 4.0 inches in length<br>- 65 percent of the smallest dimension across the panel | Damage that is a maximum of:<br>- 4.0 inches in length<br>- 65 percent of the smallest dimension across the panel | There are no limits on the dimensions of the repair |
| REPAIR PROCEDURES | ——— | SRM 51-70-04 and Paragraphs 4.C and 4.D | SRM 51-70-05 and Paragraphs 4.C and 4.E | SRM 51-70-05 and Paragraphs 4.C and 4.F |

Table 202:

| REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 2 | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY B WET LAYUP** | **CATEGORY B PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | A repair is not permitted | A repair is not permitted | A repair is not permitted | There are no limits on the dimensions of the repair |
| REPAIR PROCEDURES | —— | —— | —— | SRM 51-70-05 and Paragraph 4.D |

Table 203:

| REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 3 | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY A WET LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | A repair is not permitted | Damage that is a maximum of:<br>- 4.0 inches in length<br>- 65 percent of the smallest dimension across the panel | A repair is not permitted | There are no limits on the dimensions of the repair |
| REPAIR PROCEDURES | —— | SRM 51-70-04 and Paragraph 4.C | —— | SRM 51-70-05 and Paragraph 4.D |

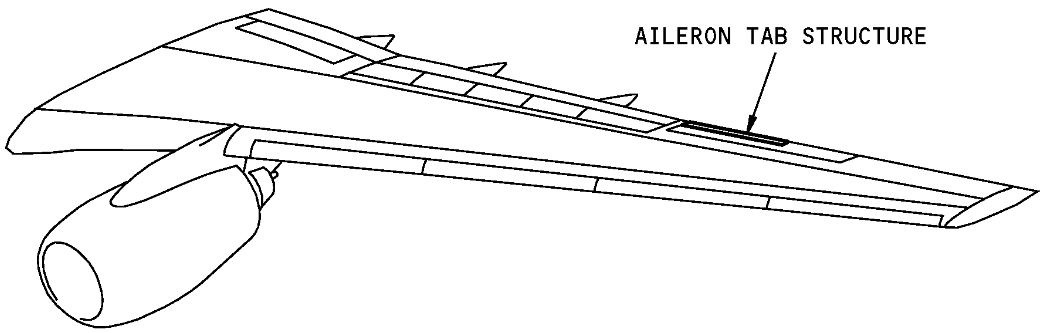
Table 204:

| REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 4 | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY B WET LAYUP** | **CATEGORY B PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | A repair is not permitted | A repair is not permitted | A repair is not permitted | There are no limits on the dimensions of the repair |
| REPAIR PROCEDURES | —— | —— | —— | SRM 51-70-05 and Paragraphs 4.C and 4.F |

**Table 205**:

| REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 5 | | | | |
|---|---|---|---|---|
| **REPAIR TYPE** | **CATEGORY B WET LAYUP** | **CATEGORY B WET LAYUP** | **CATEGORY B PREIMPREGNATED LAYUP** | **CATEGORY A PREIMPREGNATED LAYUP** |
| **REPAIR CURE TEMPERATURE** | **150°F (66°C)** | **200°F (93°C)** | **250°F (121°C)** | **350°F (177°C)** |
| REPAIR SIZE AND LIMITS | A repair is not permitted | A repair is not permitted | A repair is not permitted | There are no limits on the dimensions of the repair |
| REPAIR PROCEDURES | —— | —— | —— | SRM 51-70-05 and Paragraph 4.D |

## REPAIR 2 - AILERON TAB STRUCTURE

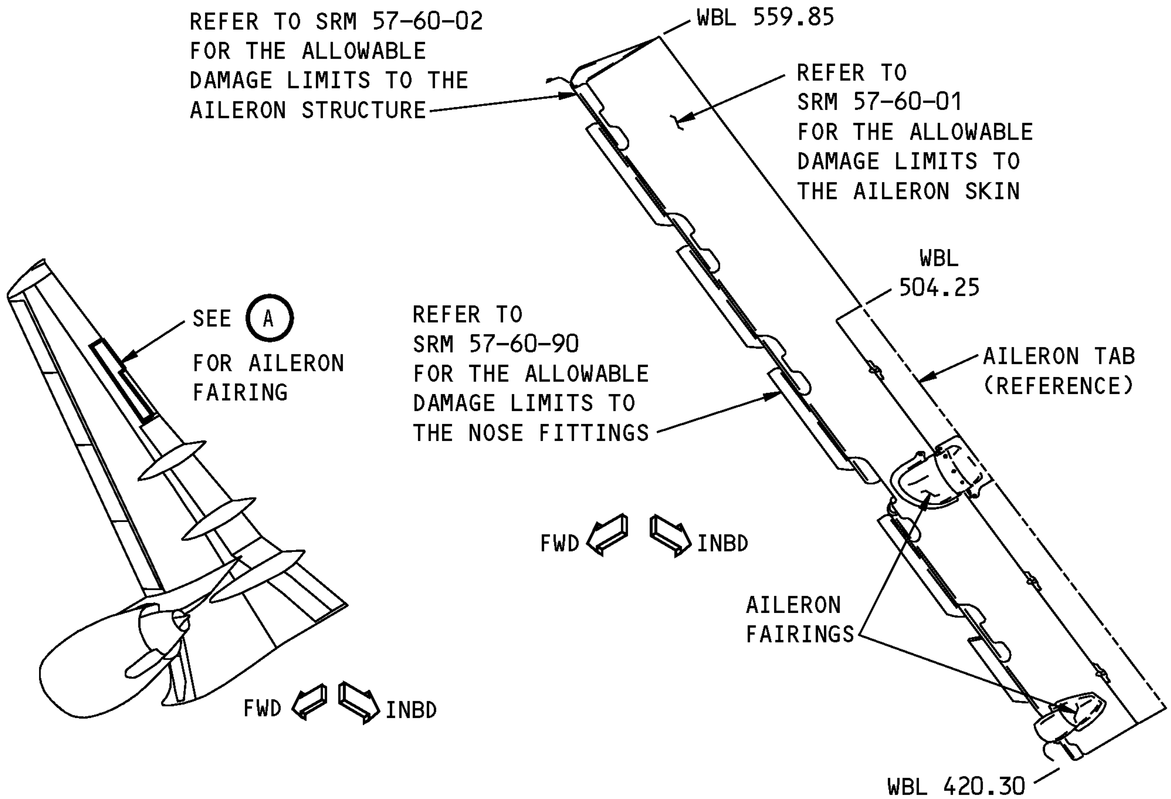



NOTE: THERE ARE NO REPAIRS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Wing Aileron Tab Structure Location**
**Figure 201**

## ALLOWABLE DAMAGE 1 - WING AILERON FAIRING



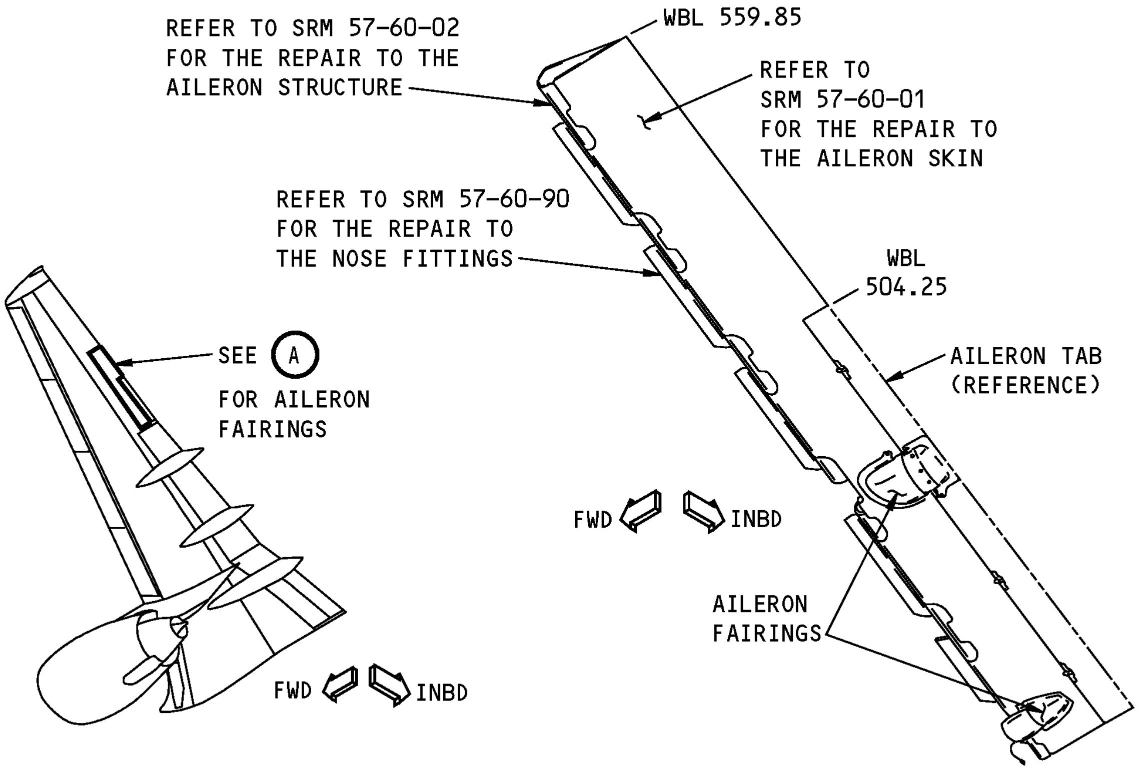

NOTE: THERE IS NO ALLOWABLE DAMAGE FOR THIS PART IN THE STRUCTURAL REPAIR MANUALAT THIS TIME.

LEFT SIDE LOWER SURFACE IS SHOWN,
RIGHT SIDE IS OPPOSITE

A

**Wing Aileron Fairing**
**Figure 101**

## REPAIR 1 - WING AILERON FAIRING



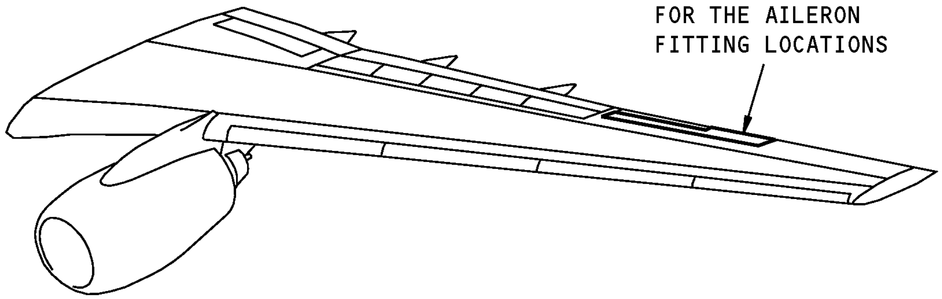

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE LOWER SURFACE IS SHOWN,
RIGHT SIDE IS OPPOSITE

A

**Wing Aileron Fairing**
**Figure 201**

## IDENTIFICATION 1 - AILERON FITTINGS



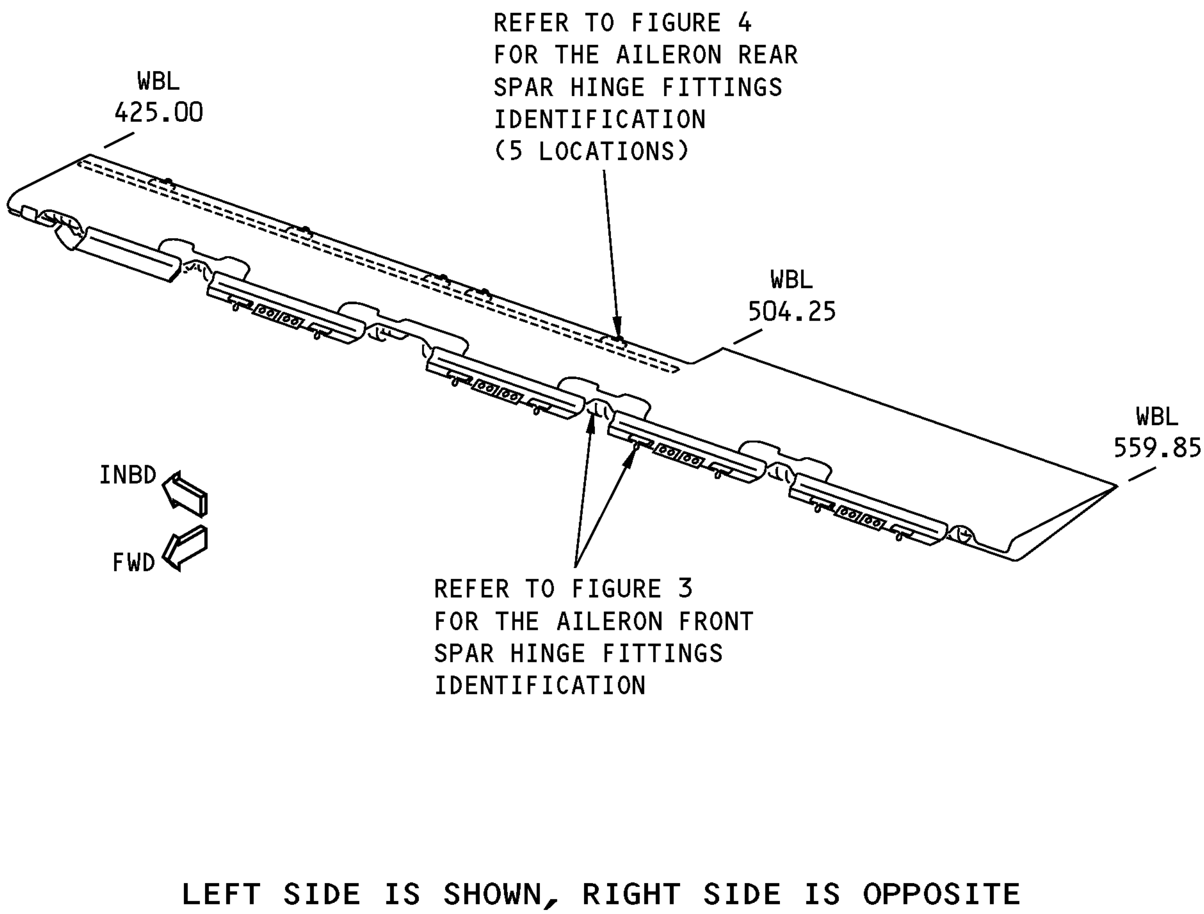

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

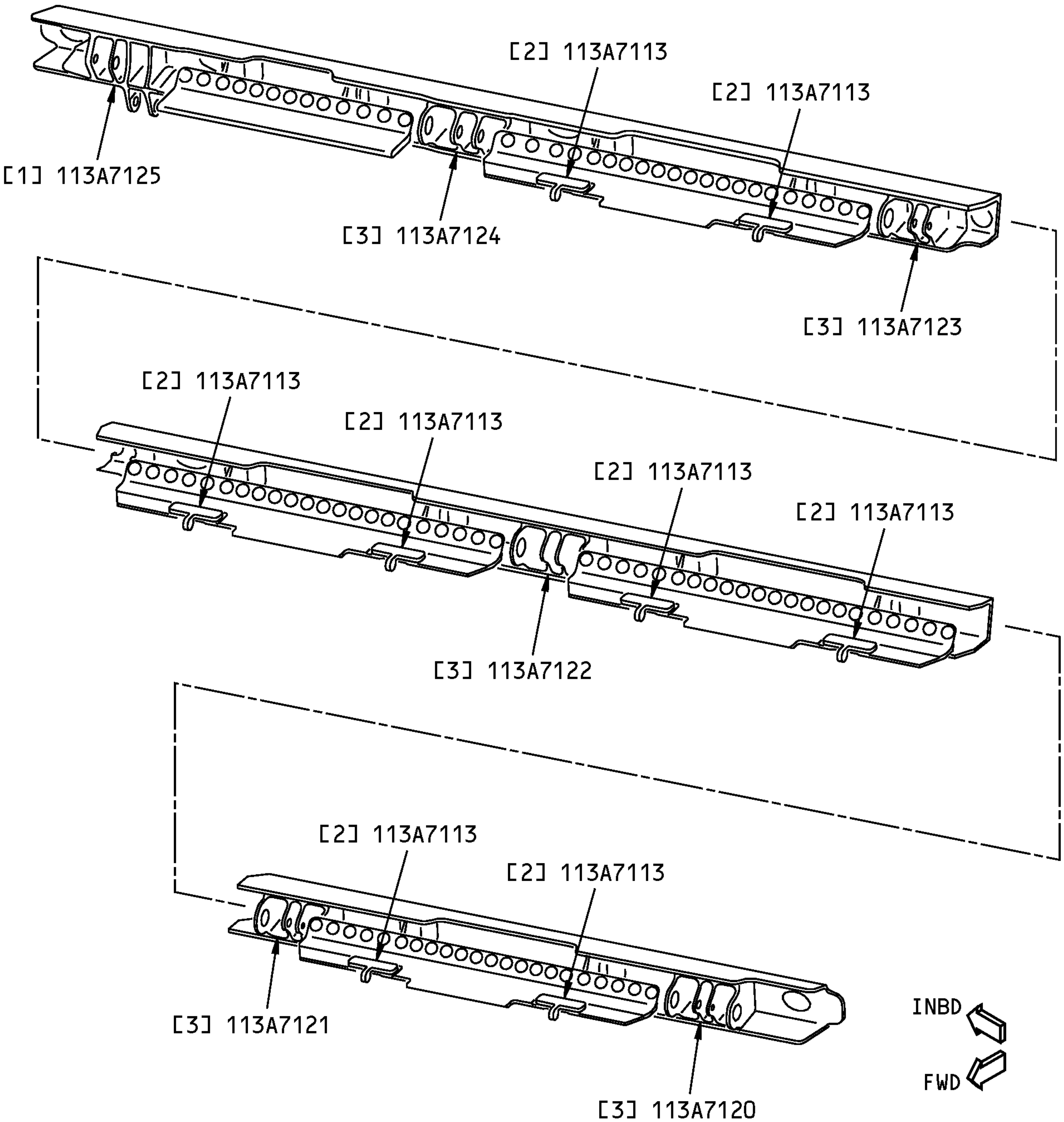
**Aileron Fitting**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 113A0003 | Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (LH) |
| 113A0004 | Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (RH) |
| 113A7001 | Aileron Installation |
| 113A7002 | Aileron/Tab Assembly |
| 113A7100 | Aileron Assembly |
| 113A7101 | Front Spar Assembly - Aileron |




**Aileron Fitting Locations**
**Figure 2**




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

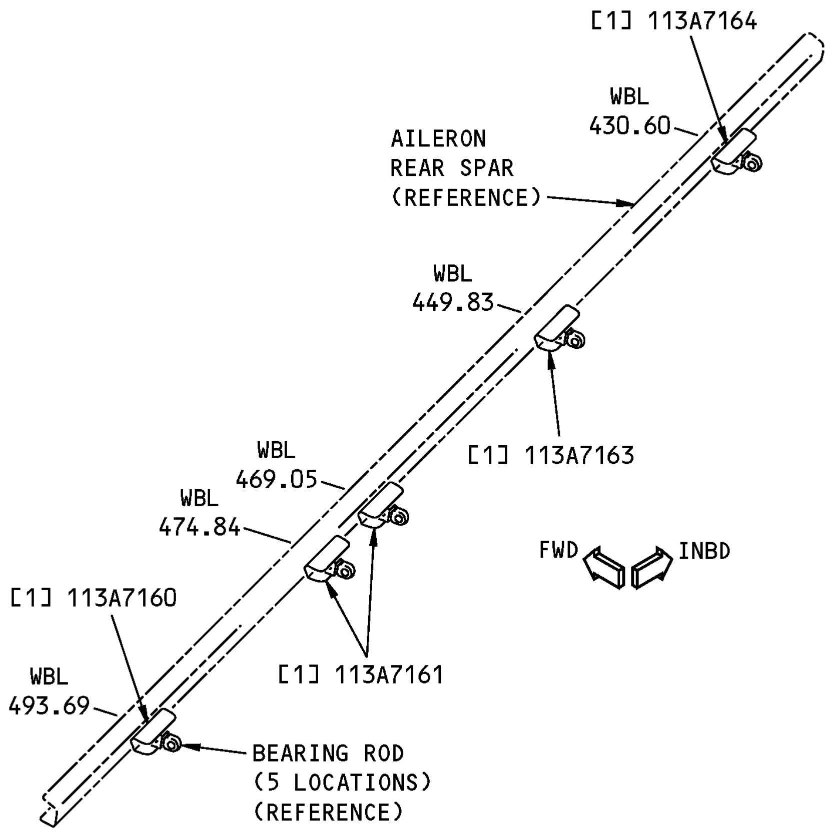
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Aileron Front Spar Hinge Fittings Identification**
**Figure 3**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 3 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Actuator Fitting | | Ti-6Al-4V titanium investment casting as given in BMS 7-310, Grade B, annealed | |
| [2] | Hinge Half (4 pairs) | | 15-5PH CRES investment casting as given in AMS 5346, heat treated to 180 KSI | |
| [3] | Hinge Fitting (Number 1 thru Number 5) | | 7050-T7451 plate as given in AMS 4050. Grain direction controlled part | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

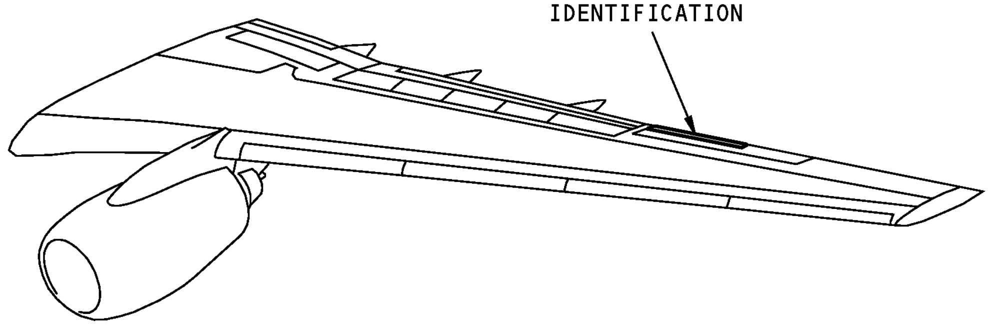
**Aileron Rear Spar Hinge Fittings Identification**
**Figure 4**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T**[*[1]] | **MATERIAL** | **EFFECTIVITY** |
| [1] | Hinge Fitting (5) | | A356.0-T6 aluminum casting as given in MIL-A-21180, Class 2 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

## IDENTIFICATION 2 - AILERON TAB FITTINGS



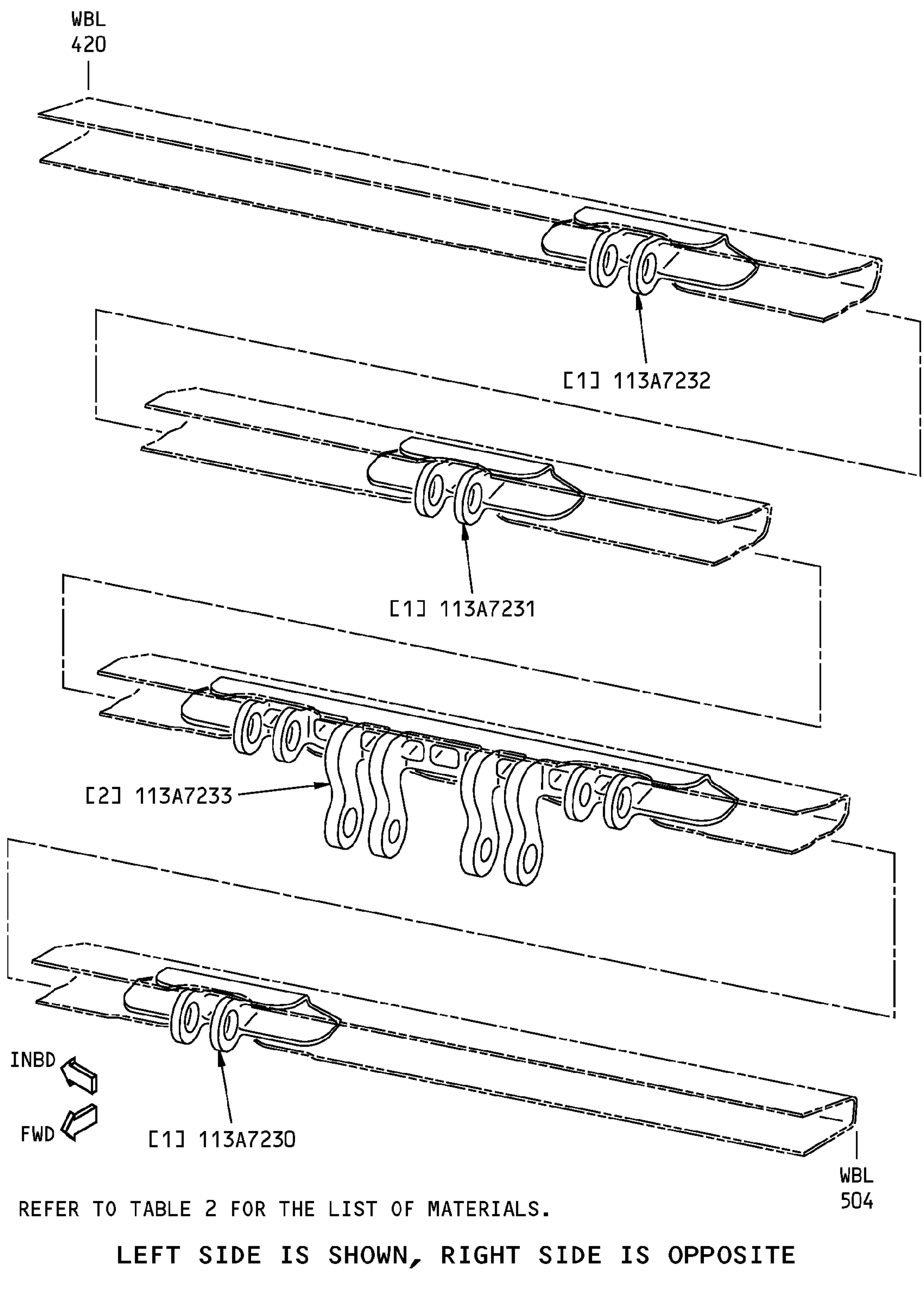

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Aileron Tab Fitting Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 113A0003 | Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (LH) |
| 113A0004 | Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (RH) |
| 113A7001 | Aileron Installation |
| 113A7002 | Aileron/Tab Assembly |
| 113A7200 | Tab Assembly - Aileron |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Aileron Tab Hinge Fitting Identification**
**Figure 2**

**Table 2**:

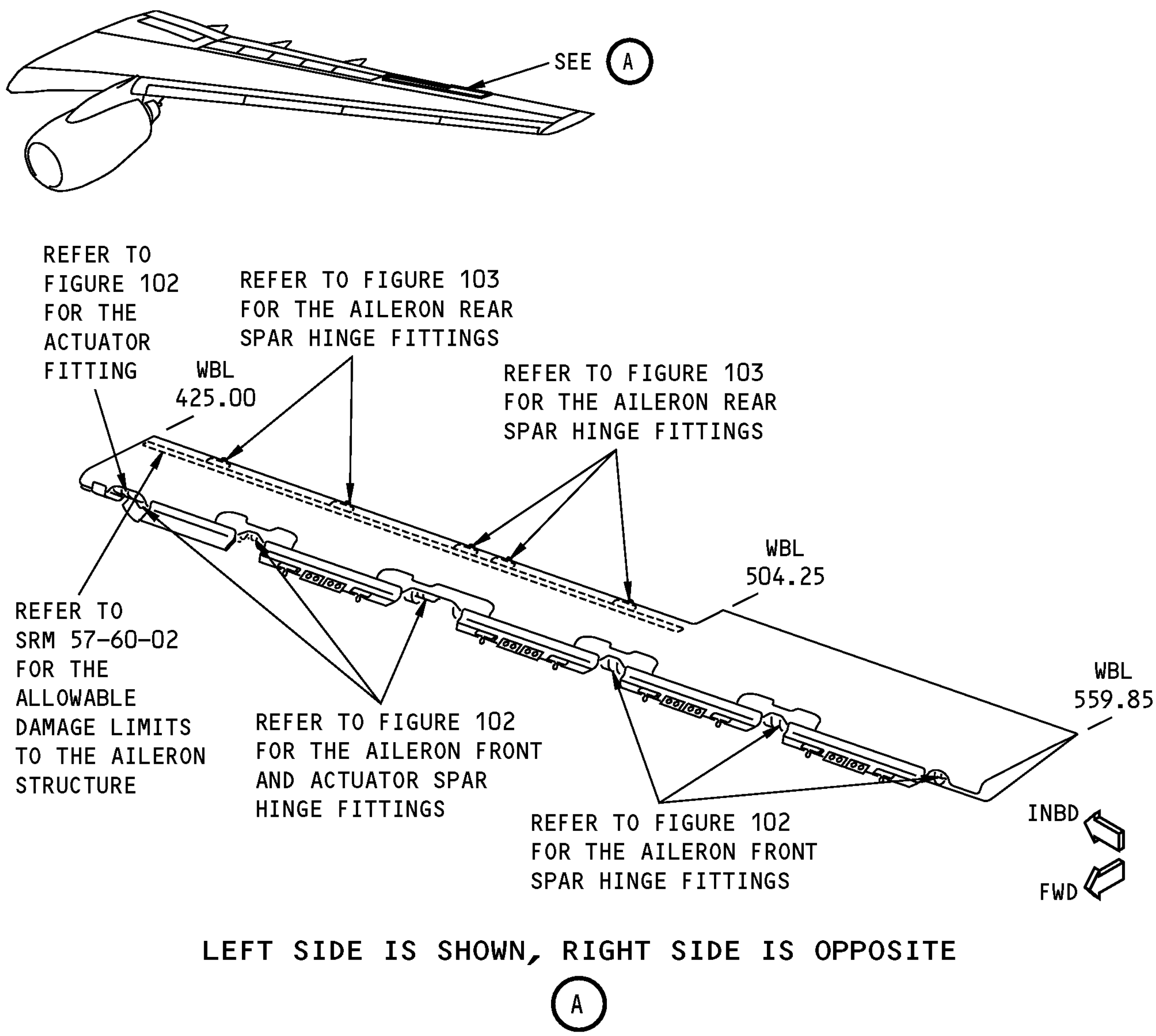
| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Tab Hinge Fitting (3) | | A356.0-T6 aluminum casting as given in MIL-A-21180, strength Class 2 | |
| [2] | Tab Actuator Fitting | | A356.0-T6 aluminum casting as given in MIL-A-21180, strength Class 2 | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

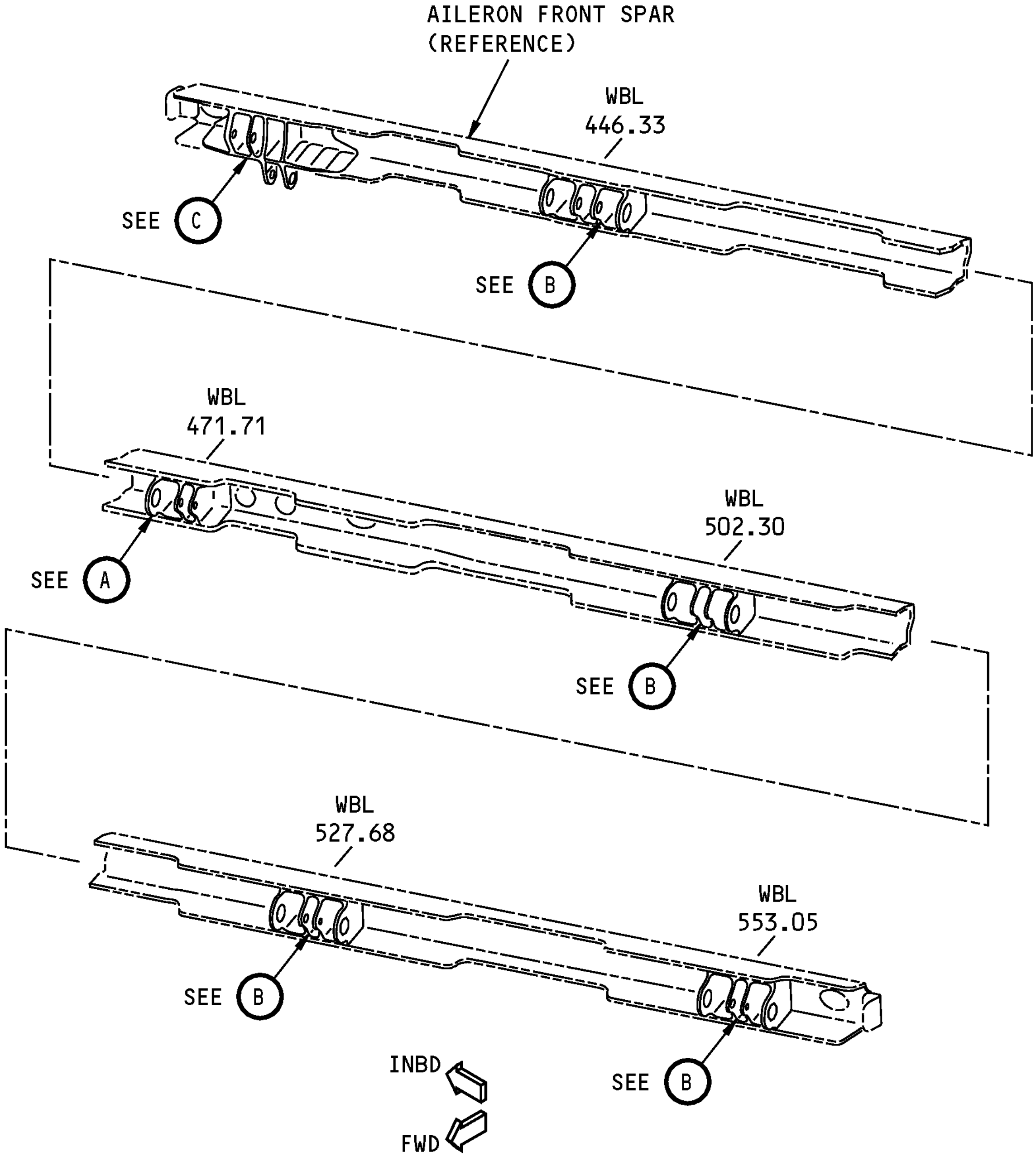
## ALLOWABLE DAMAGE 1 - AILERON HINGE FITTINGS

### 1. Applicability

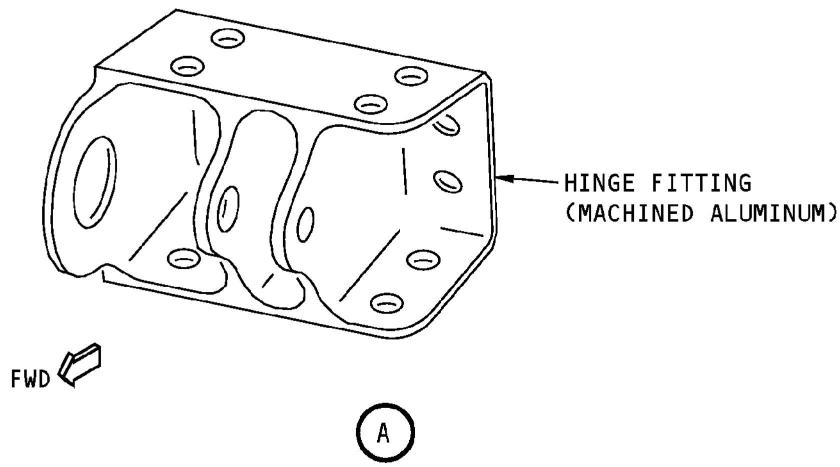
A. This subject gives the allowable damage limits for the Wing Aileron Hinge Fittings shown in Aileron Fitting Locations, Figure 101/ALLOWABLE DAMAGE 1.

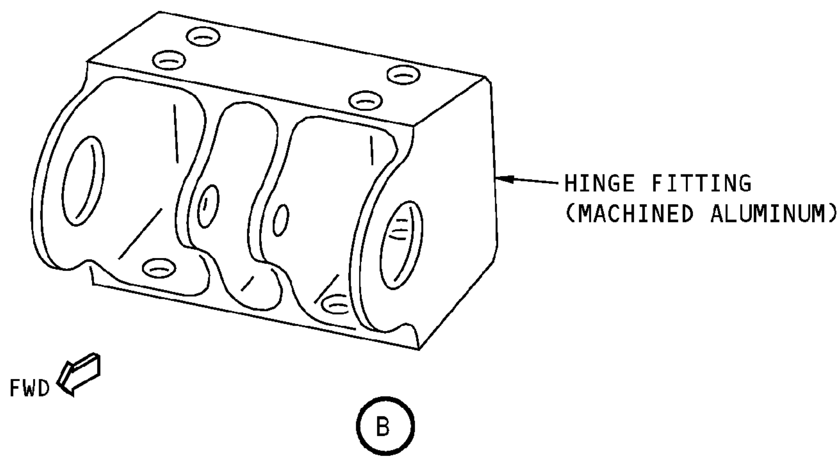


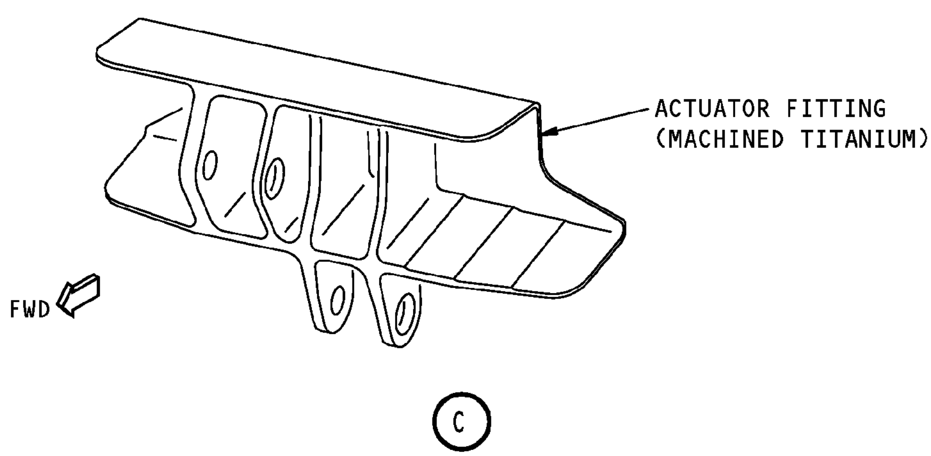

**Aileron Fitting Locations**
**Figure 101**



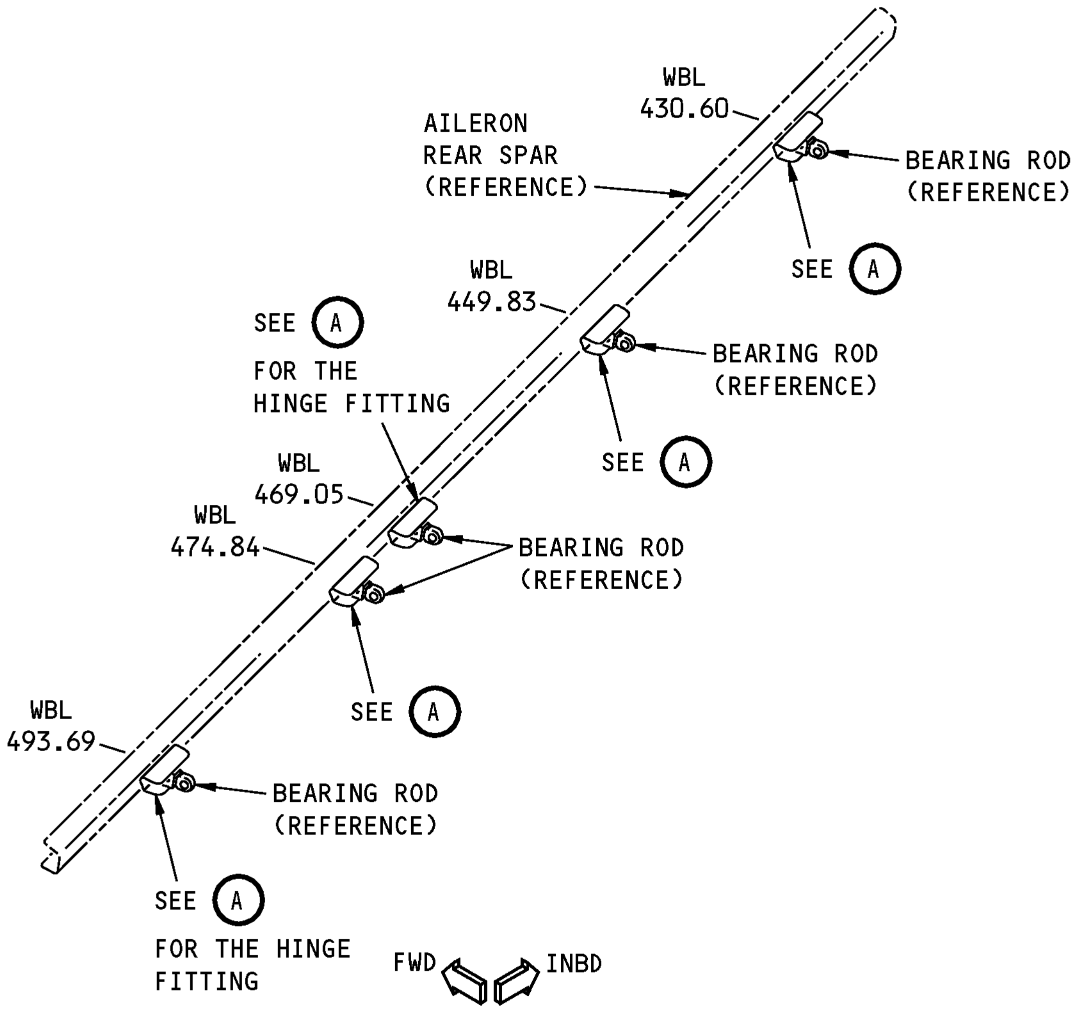

**Aileron Front Spar Fitting**
**Figure 102 (Sheet 1 of 2)**



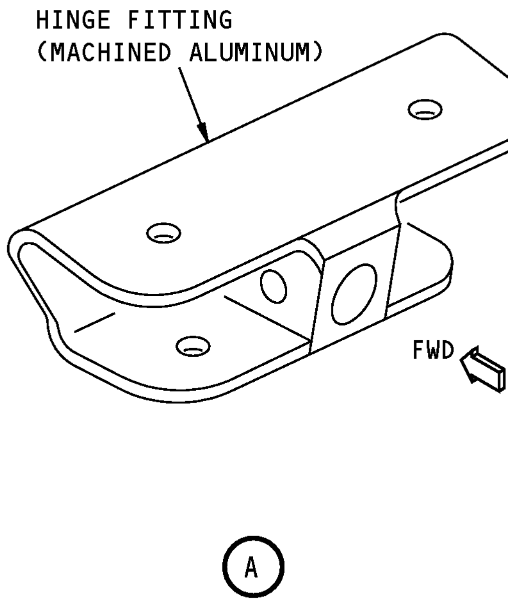




**Aileron Front Spar Fitting**
**Figure 102 (Sheet 2 of 2)**




**Aileron Rear Spar Hinge Fitting**
**Figure 103 (Sheet 1 of 2)**




**Aileron Rear Spar Hinge Fitting**
**Figure 103 (Sheet 2 of 2)**

## 2. General

A. Do the steps that follow if you have damage.

(1) Remove the damage. Refer to 51-10-02 for the investigation and cleanup procedures.

(2) Refer to 51-30-03 for the sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for the sources of the equipment and tools you can use to remove the damage.

B. Do the steps that follow after you remove the damage.

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the parts, but not the inner surfaces of the lug bores.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| AMM 51-21-99 | DECORATIVE EXTERIOR PAINT SYSTEM |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

## 4. Allowable Damage Limits

A. Hinge Fittings (Machined Aluminum)

(1) Cracks:

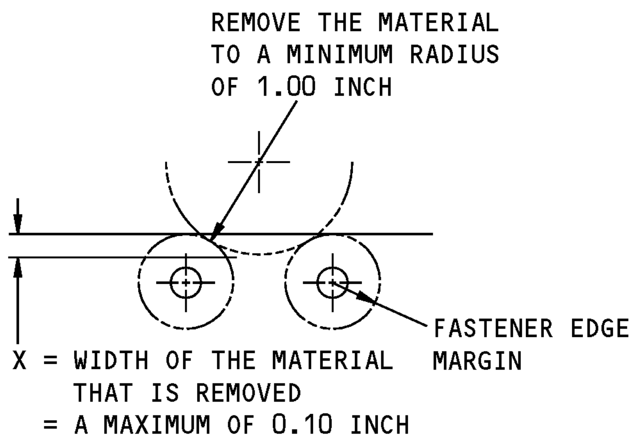
(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , , D , and E .

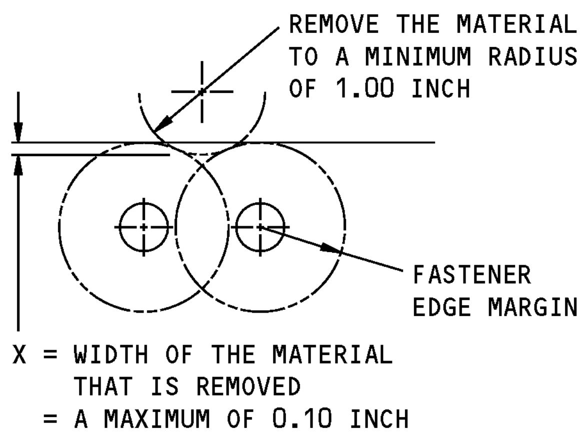
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

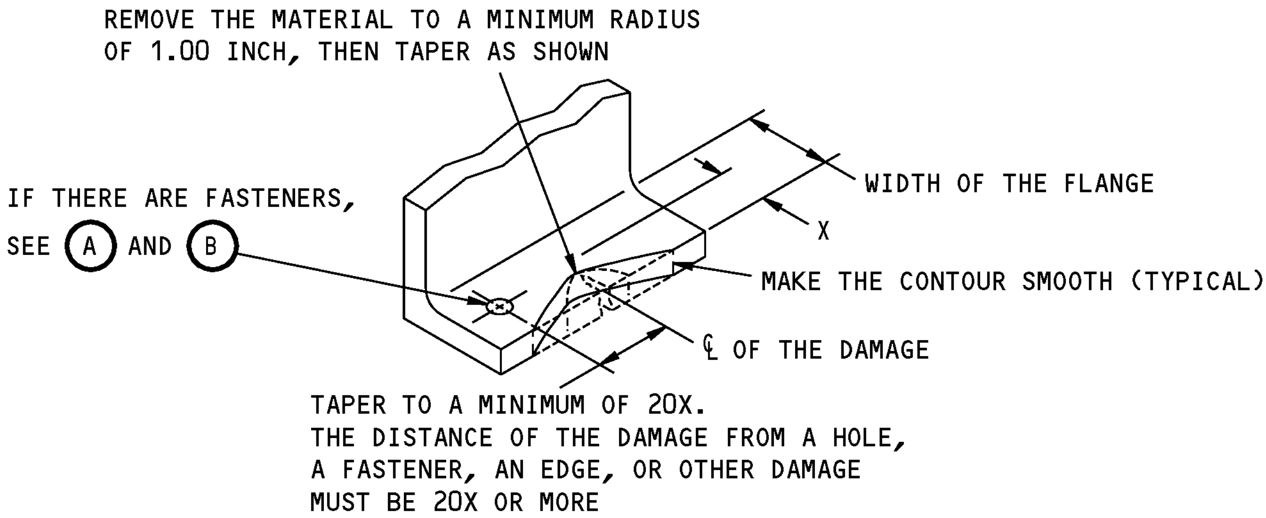
A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

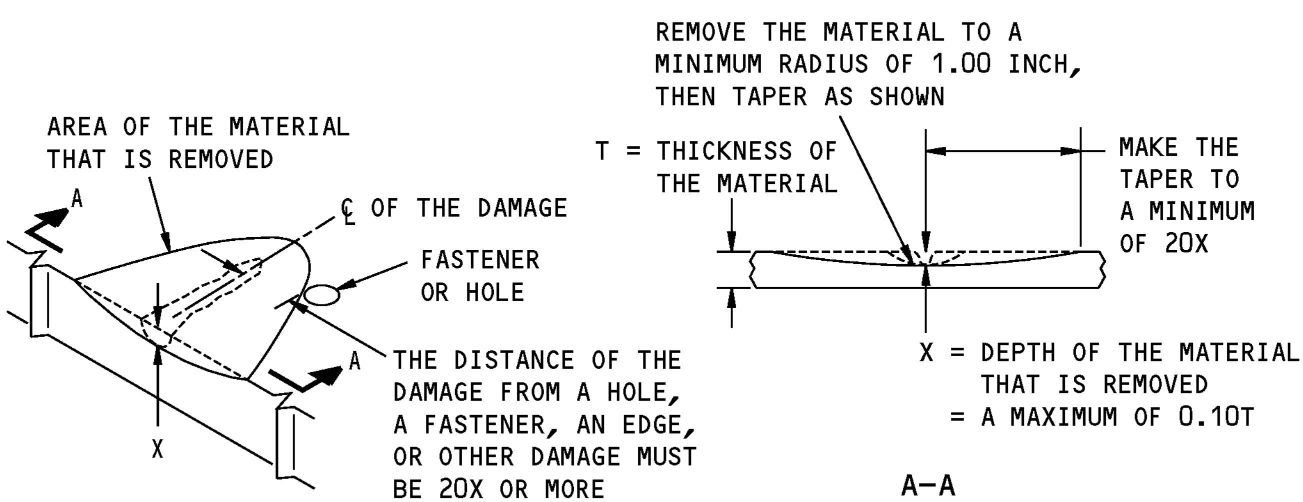
**Allowable Damage Limits**
**Figure 104 (Sheet 1 of 3)**




X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

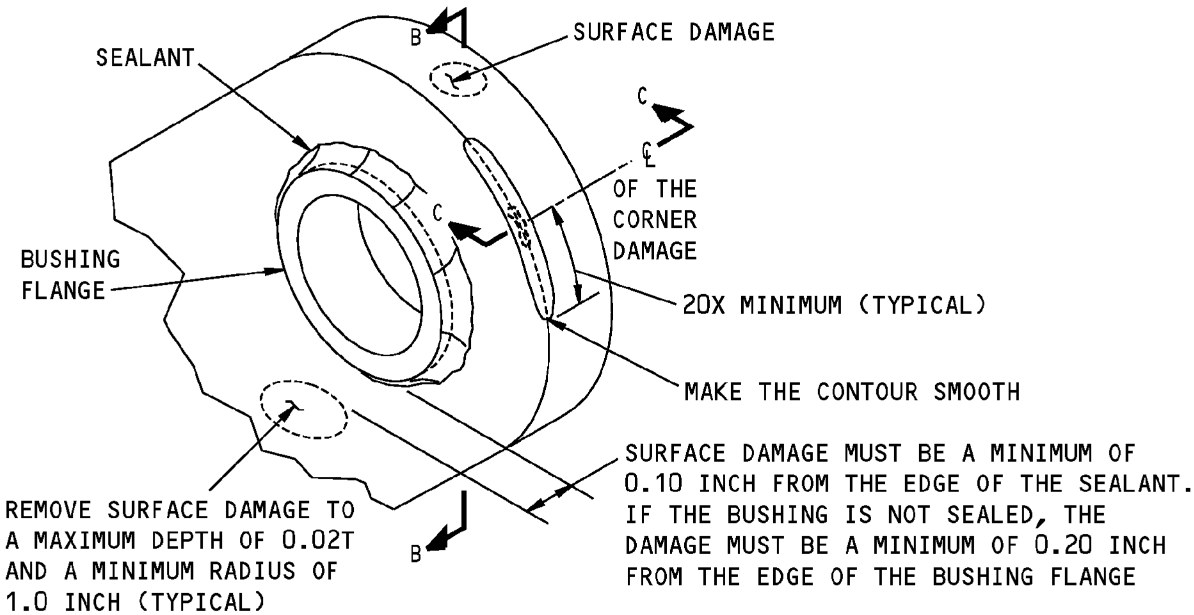
C




REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

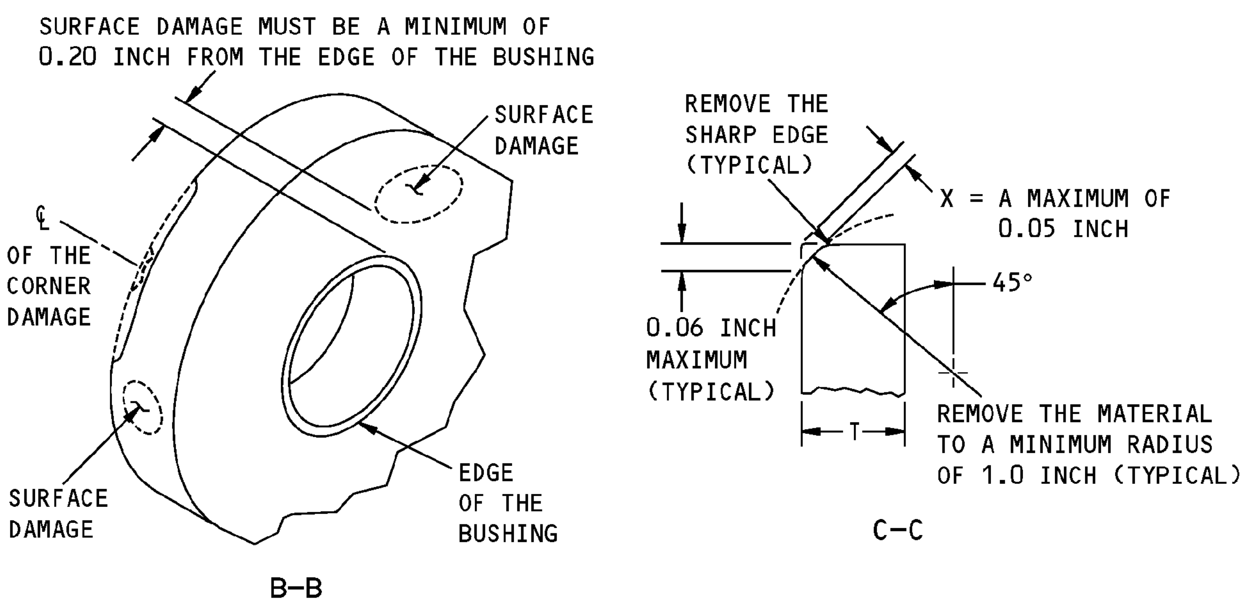
**Allowable Damage Limits**
**Figure 104 (Sheet 2 of 3)**




NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

E

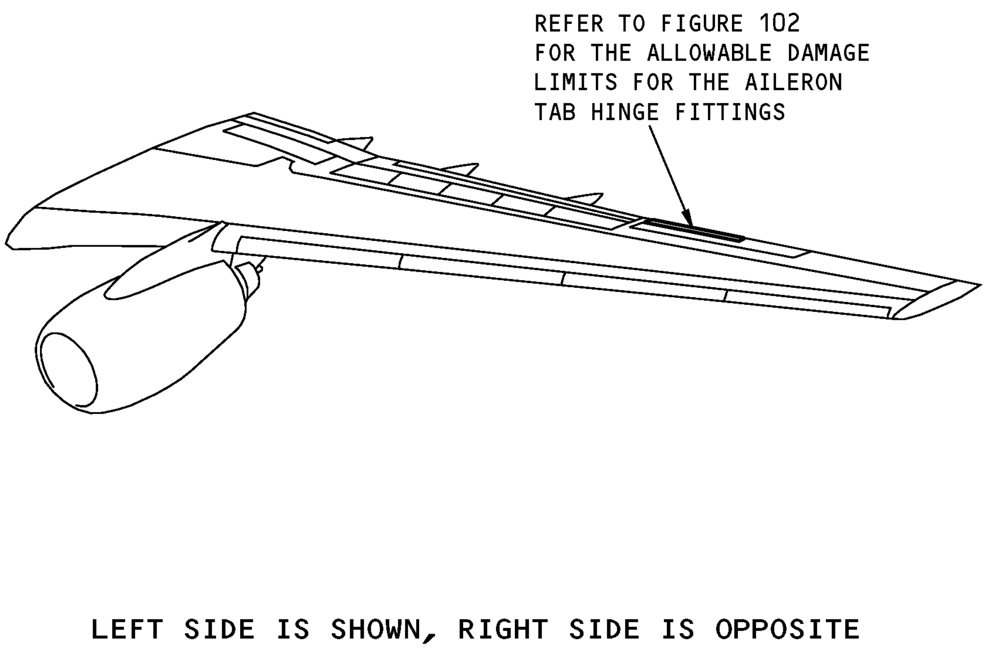



**Allowable Damage Limits**
**Figure 104 (Sheet 3 of 3)**

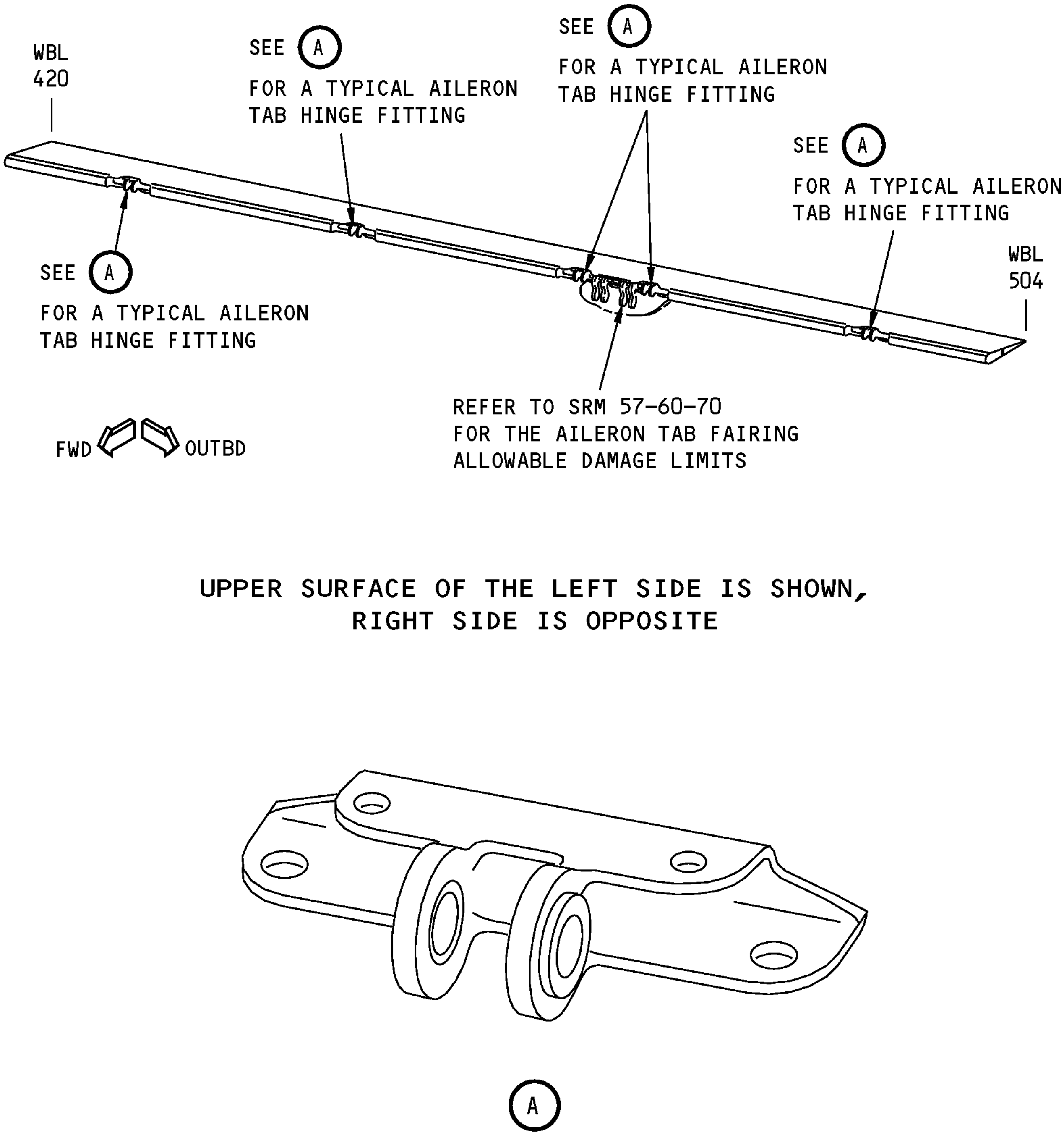
## ALLOWABLE DAMAGE 2 - AILERON TAB HINGE FITTINGS

1. **Applicability**

   A. This subject gives the allowable damage limits for the aileron tab hinge fittings shown in Aileron Tab Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.




**Aileron Tab Skin Panel Location**
**Figure 101**




**Aileron Tab Hinge Fitting**
**Figure 102**

2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tolls you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

**WARNING:** MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the bullnose links, but not the inner surfaces of the link holes.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply a layer of BMS 10-79, Type III primer to the reworked areas. Refer to SOPM 20-44-04.

3. **References**

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-06 | SHOT PEENING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| SOPM 20-10-03 | General - Shot Peening Procedures |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |

4. **Allowable Damage Limits**

A. Hinge Fittings

(1) Cracks:

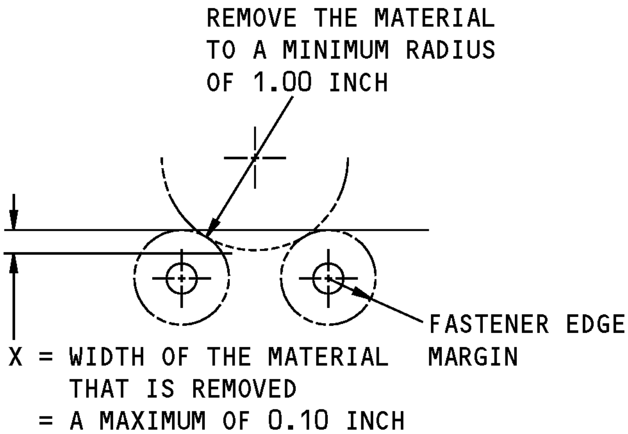
(a) Remove the damage as shown in Aileron Tab Hinge Fitting, Figure 102/ALLOWABLE DAMAGE 2, Details A and .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Aileron Tab Hinge Fitting, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and D .

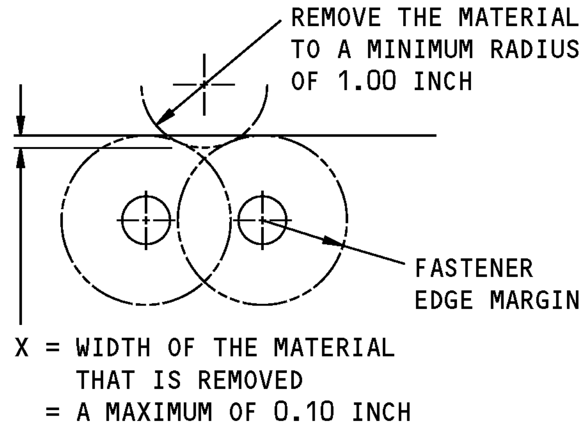
(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.



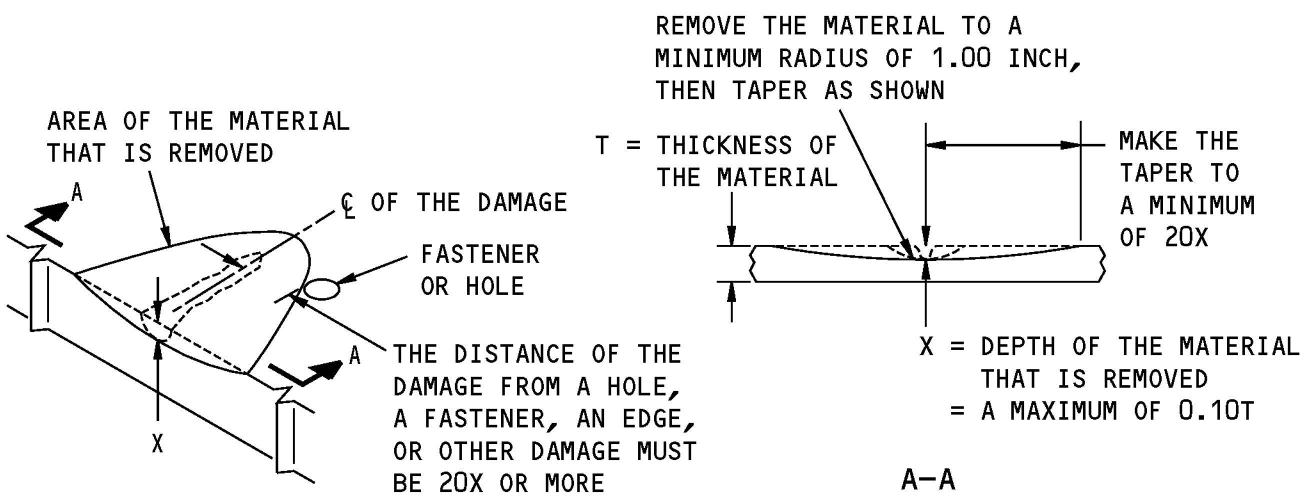

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

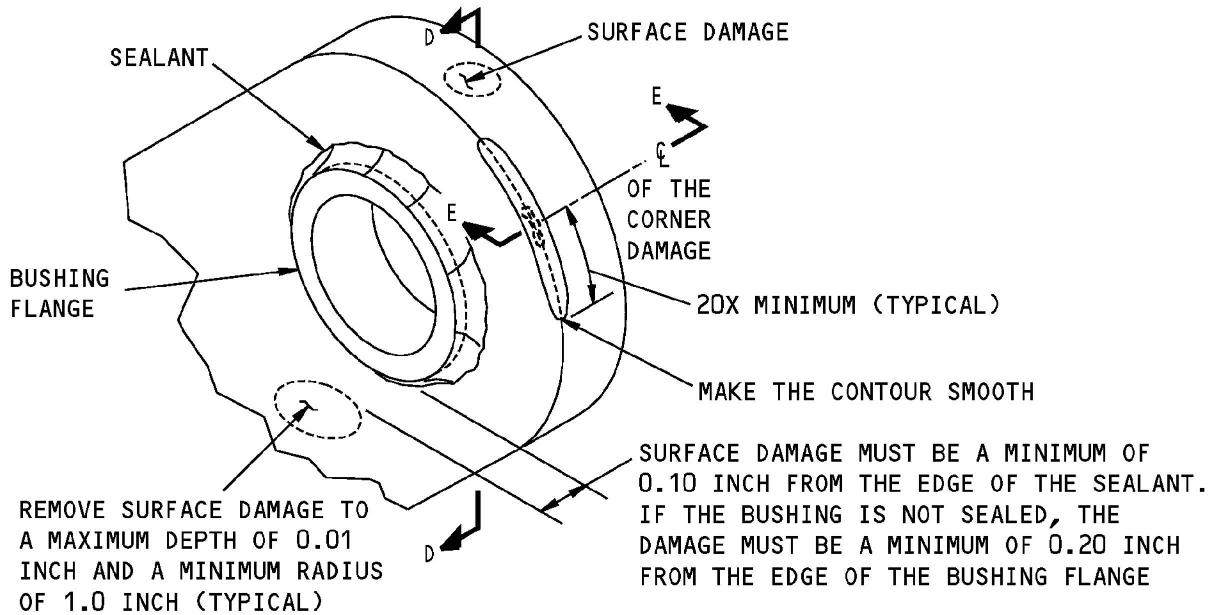
B




REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

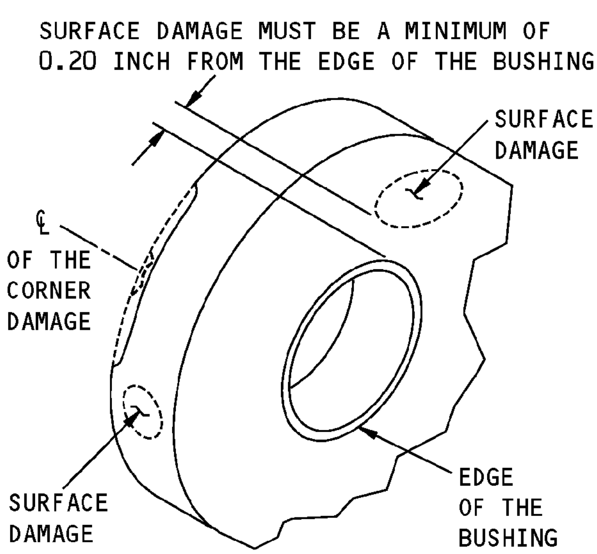
**Allowable Damage Limits**
**Figure 103 (Sheet 1 of 2)**



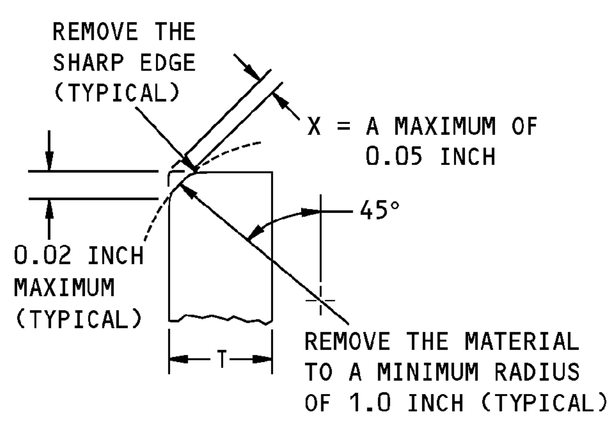

NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

D



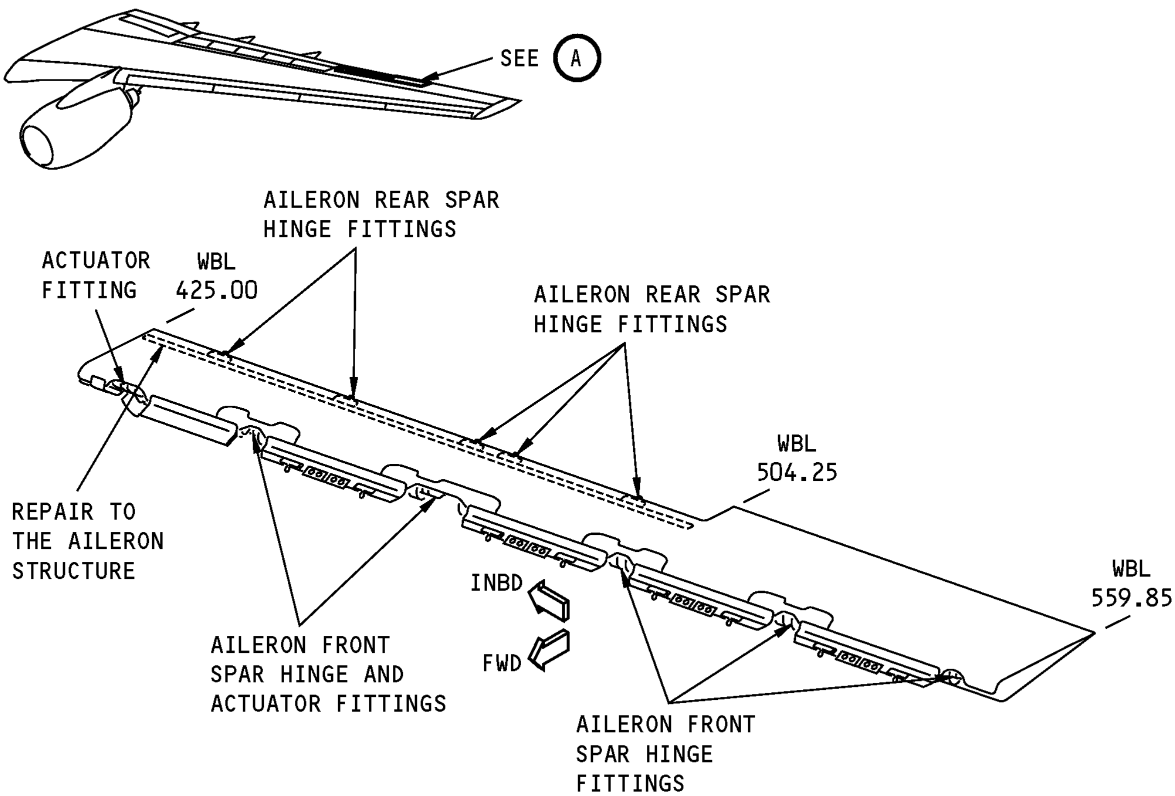

D-D




E-E

**Allowable Damage Limits**
**Figure 103 (Sheet 2 of 2)**

## REPAIR 1 - AILERON HINGE FITTINGS



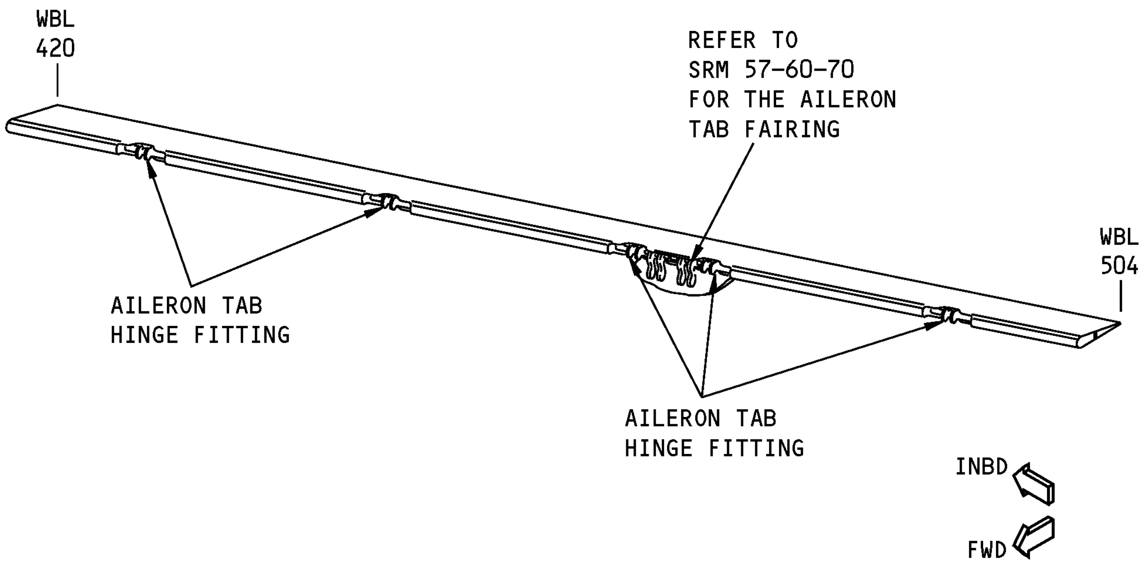

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

**Aileron Fitting Locations**
**Figure 201**

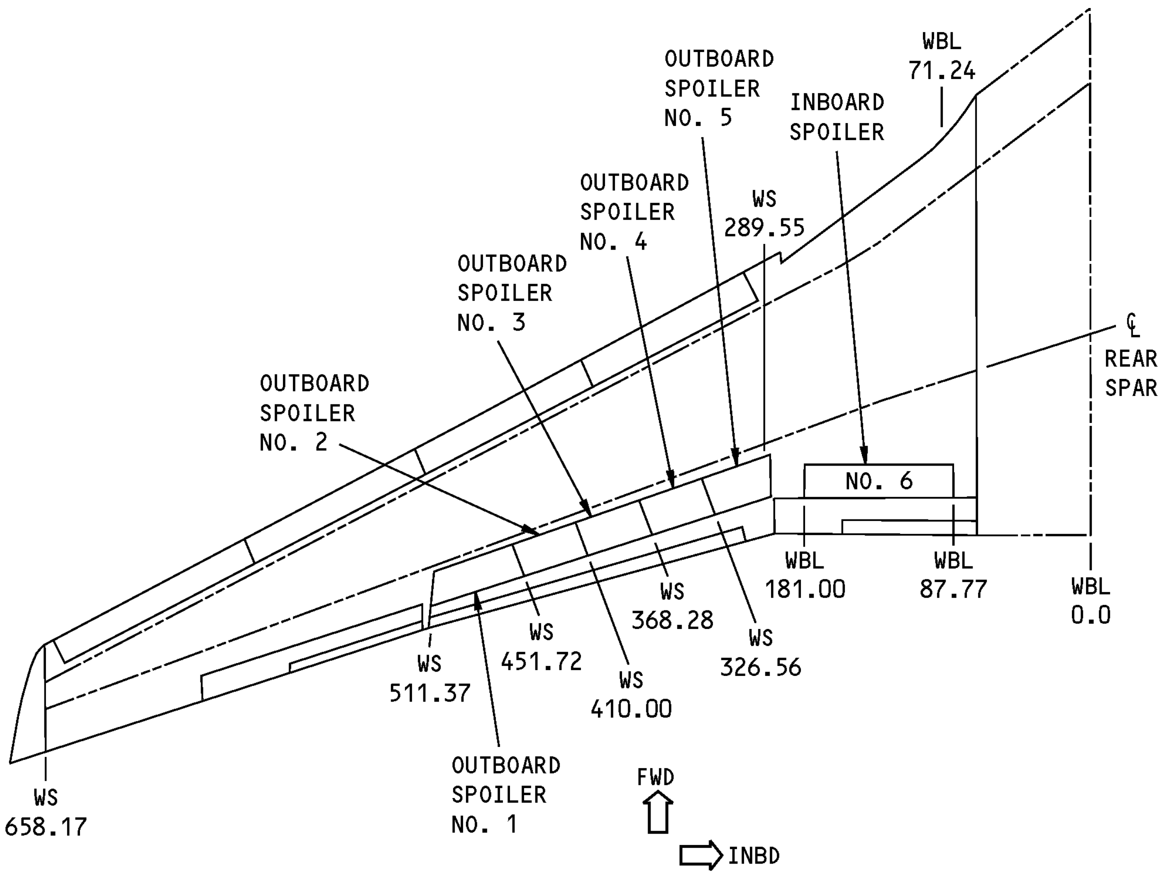
## REPAIR 2 - AILERON TAB HINGE FITTINGS




NOTE: THERE ARE NO REPAIRS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

**Aileron Tab Hinge Fitting Repair**
**Figure 201**

## GENERAL - SPOILER DIAGRAM



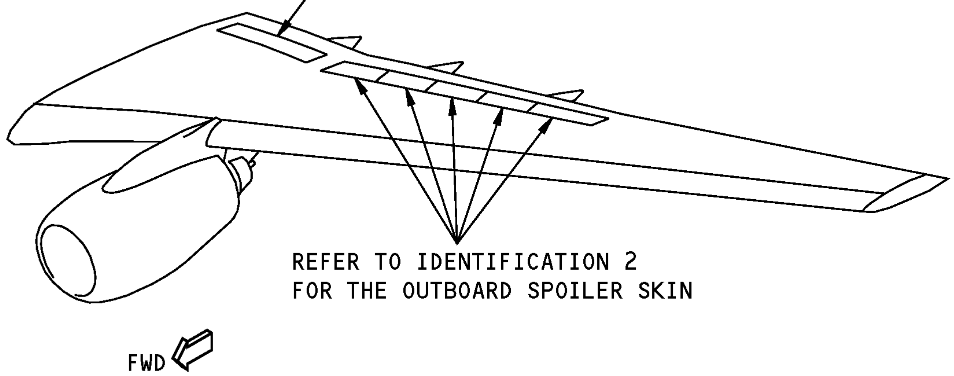

NOTE: FOR TRAILING EDGE, SEE SRM 57-51-00
FOR TRAILING EDGE FLAPS, SEE SRM 57-53-00
FOR AILERONS, SEE SRM 57-60-00

**Spoiler Diagram**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 110A1502 | Structures Diagram, Wing - Model 737X |
| 113A4000 | Drawing Index - Spoilers |
| 113A4001 | Spoiler Installation |

## IDENTIFICATION 1 - INBOARD SPOILER SKIN



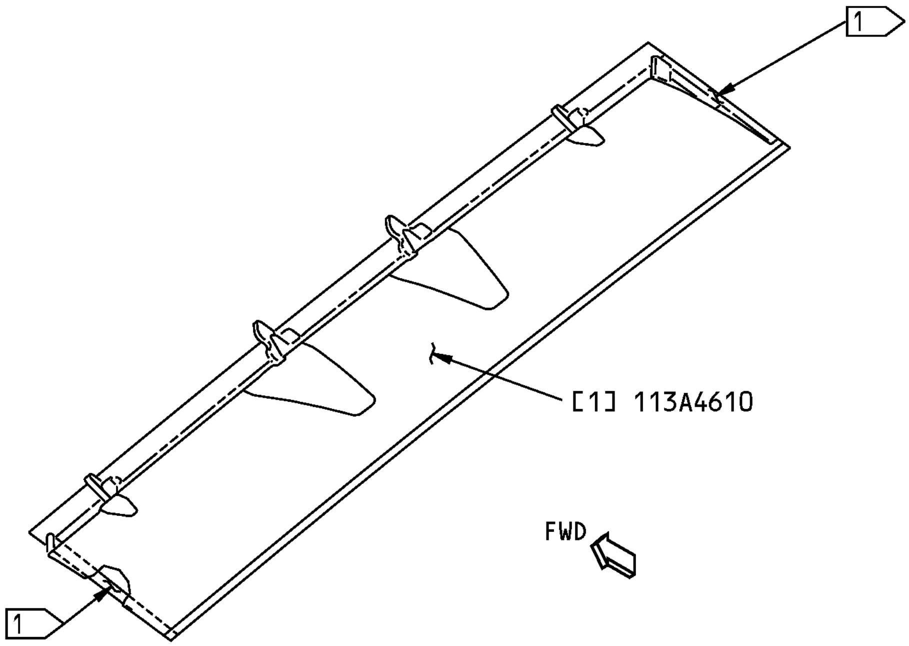

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Inboard Spoiler Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A4001 | Spoiler Installation |
| 113A4600 | Spoiler Assembly - 6 and 7, Inboard Ground |




NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

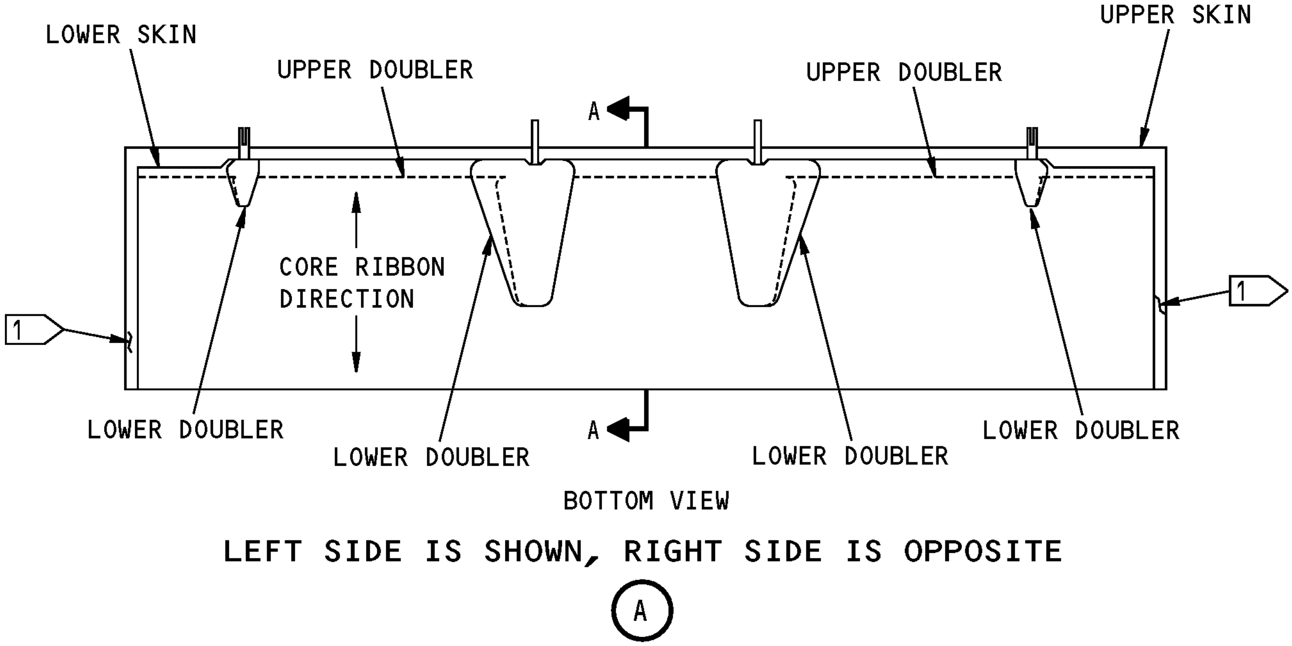
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

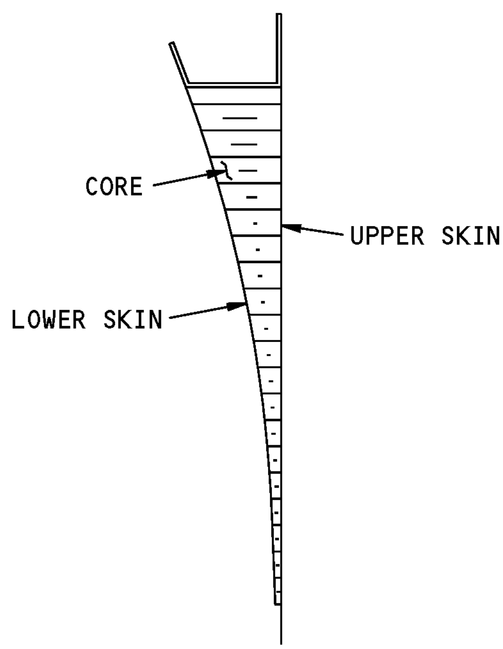
**Inboard Spoiler Number 6 and 7 Skin Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Spoiler Assembly - Bonded Panel | | | |
| | Skin - Upper | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - lower | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Doubler - Upper | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Doubler - lower | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 3 for the core ribbon direction | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).



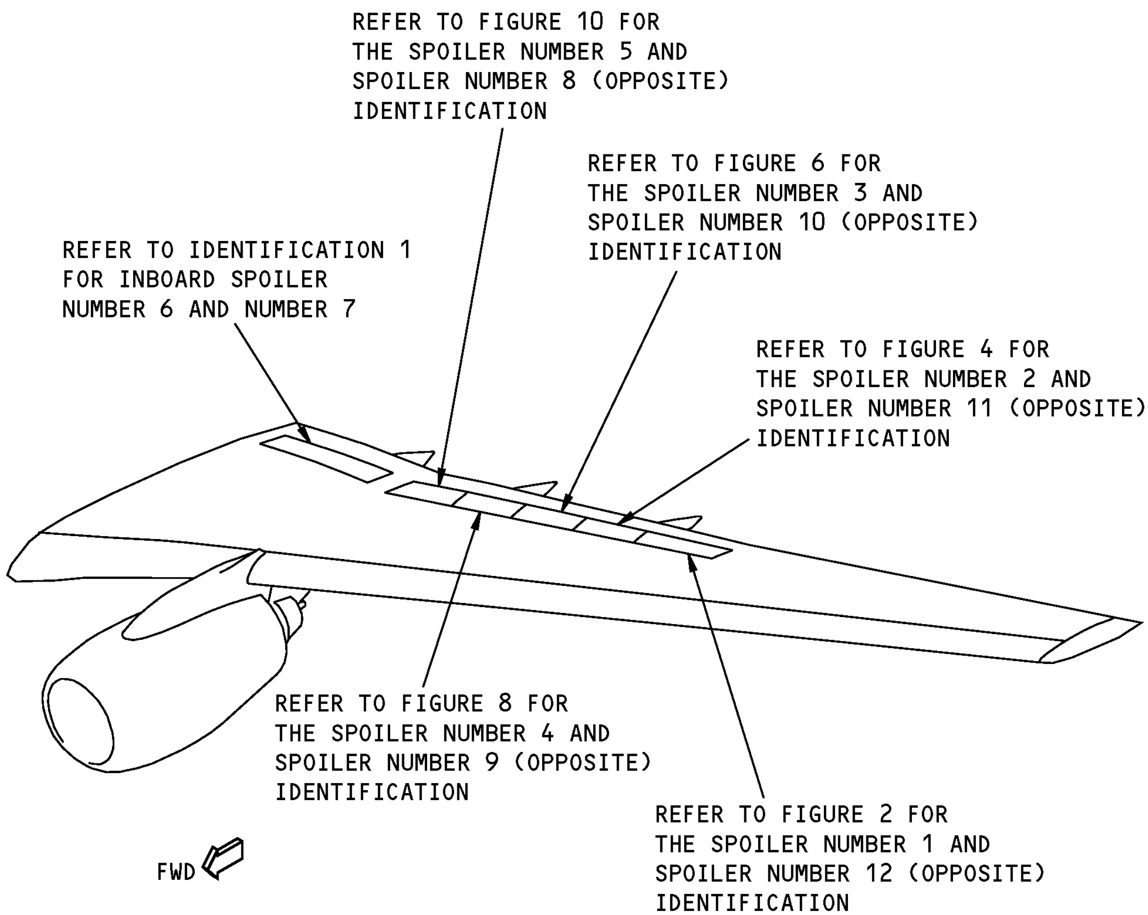




NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

**Core Ribbon Direction for Figure 2, Item [1]**
**Figure 3**

## IDENTIFICATION 2 - OUTBOARD SPOILER SKINS




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

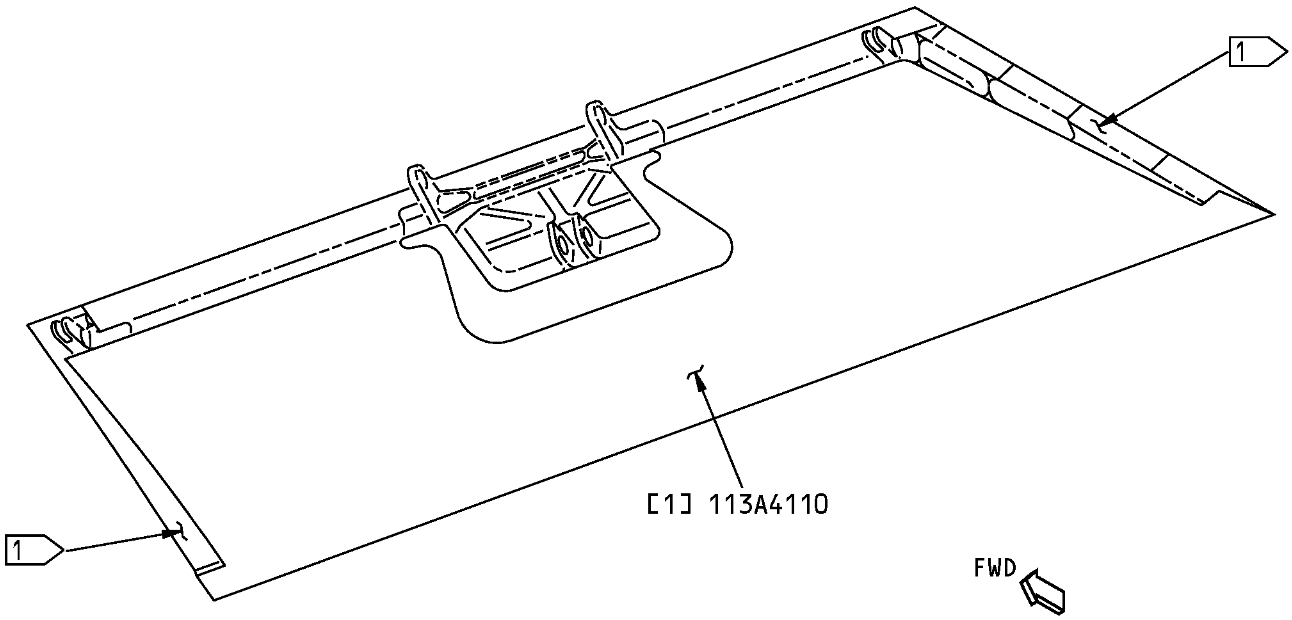
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outboard Spoiler Location**
**Figure 1**

**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A4001 | Spoiler Installation |
| 113A4100 | Spoiler Assembly - 1 and 12, Outboard Ground |
| 113A4110 | Panel Bond Assembly - Spoiler 1 and 12 |
| 113A4200 | Spoiler Assembly - 2 and 11, Flight |
| 113A4210 | Panel Bond Assembly - Spoiler 2 and 11 |
| 113A4300 | Spoiler Assembly - 3 and 10, Flight |

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A4310 | Panel Bond Assembly - Spoiler 3 and 10 |
| 113A4400 | Spoiler Assembly - 4 and 9, Flight |
| 113A4410 | Panel Bond Assembly - Spoiler 4 and 9 |
| 113A4500 | Spoiler Assembly - 5 and 8, Flight |
| 113A4510 | Panel Bond Assembly - Spoiler 5 and 8 |




NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

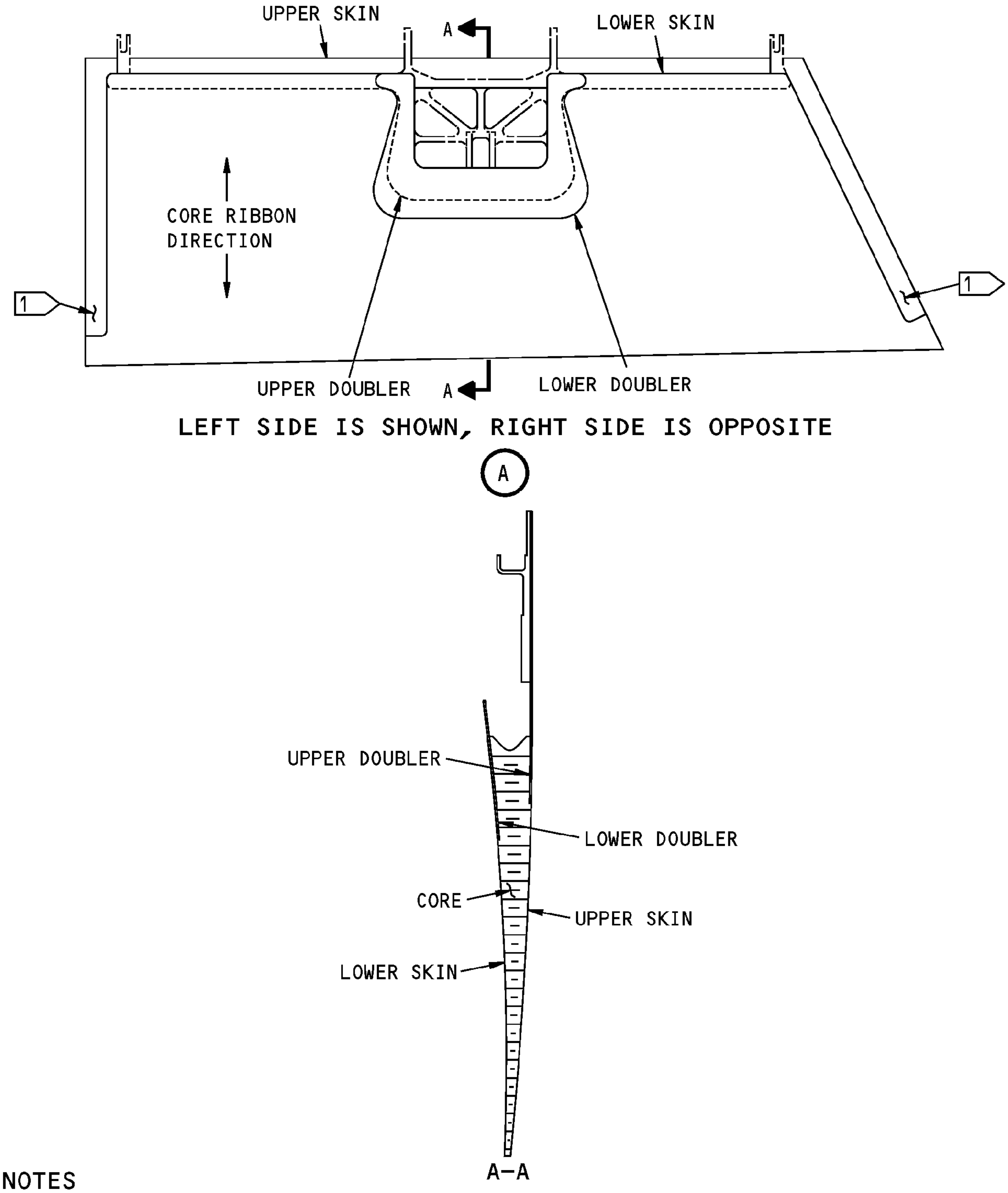
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outboard Spoiler No. 1 and 12 Skin Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T*[1] | MATERIAL | EFFECTIVITY |
| [1] | Spoiler Assembly, Numbers 1 and 12 Bonded Panel | | | |
| | Doubler - Lower | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Doubler - Upper | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - Lower | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - Upper | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 3 for the core ribbon direction | |

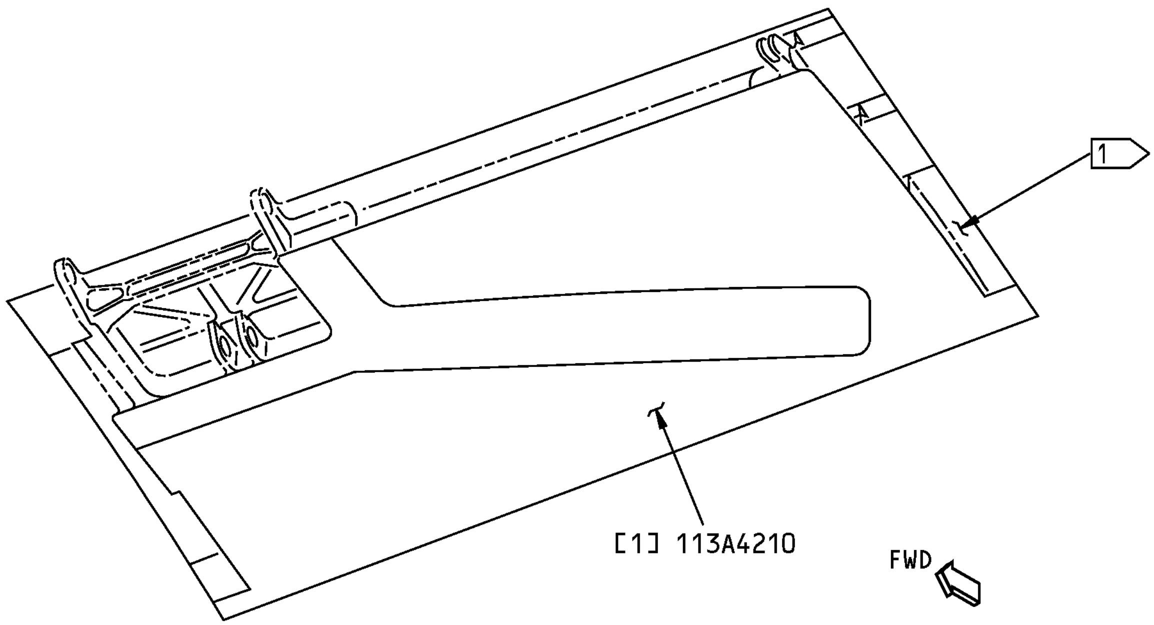
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

**Core Ribbon Direction for Figure 2, Item [1]**
**Figure 3**




NOTES

- REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

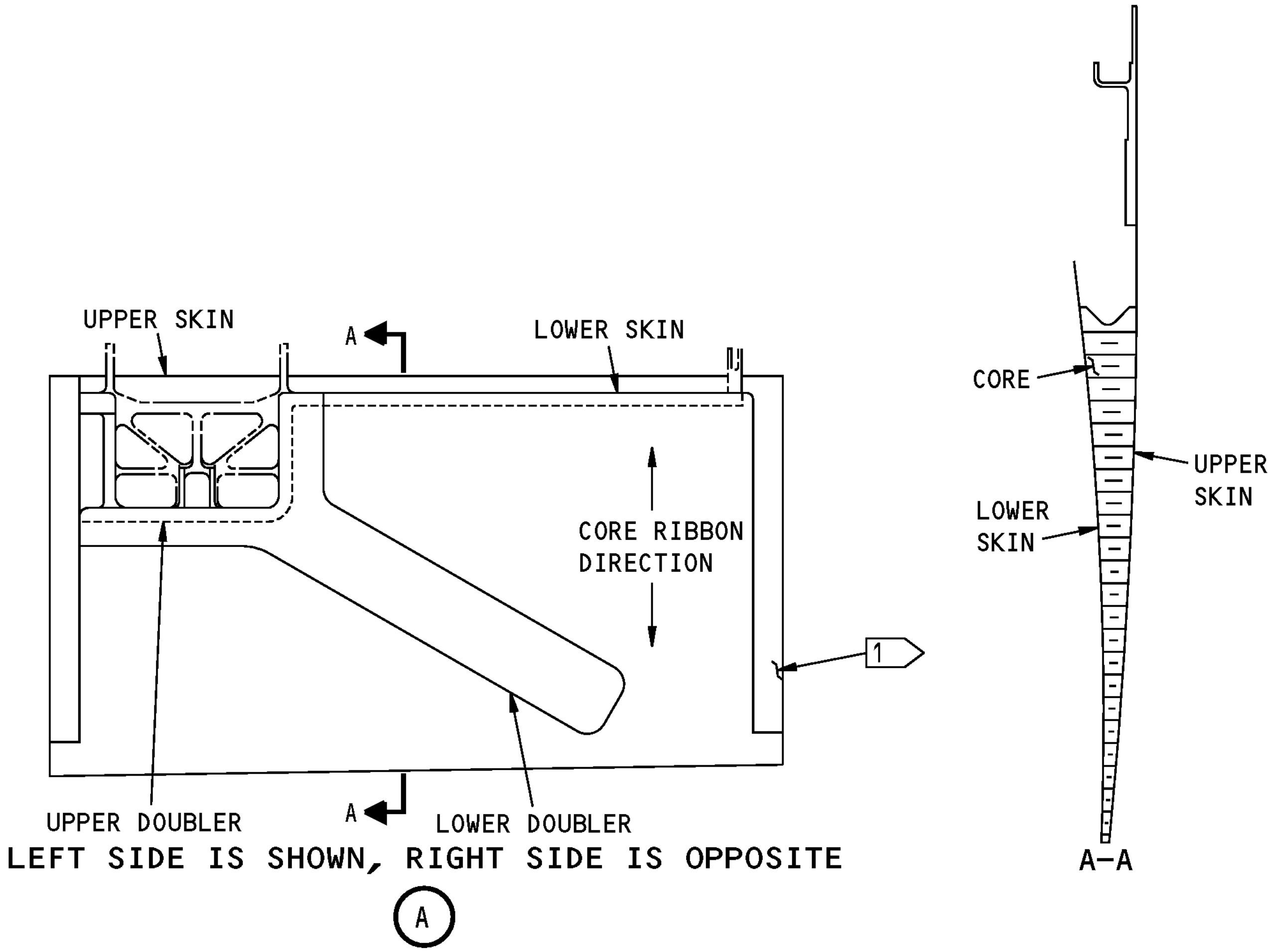
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outboard Spoiler Number 2 and 11 Skin Identification**
**Figure 4**

**Table 3**:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Spoiler Assembly, Number 2 and 11 Bonded Panel | | | |
| | Doubler - lower | 0.032 (0.812) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Doubler - Upper | 0.032 (0.812) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - lower | 0.032 (0.812) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - Upper | 0.032 (0.812) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 5 for the core ribbon direction | |

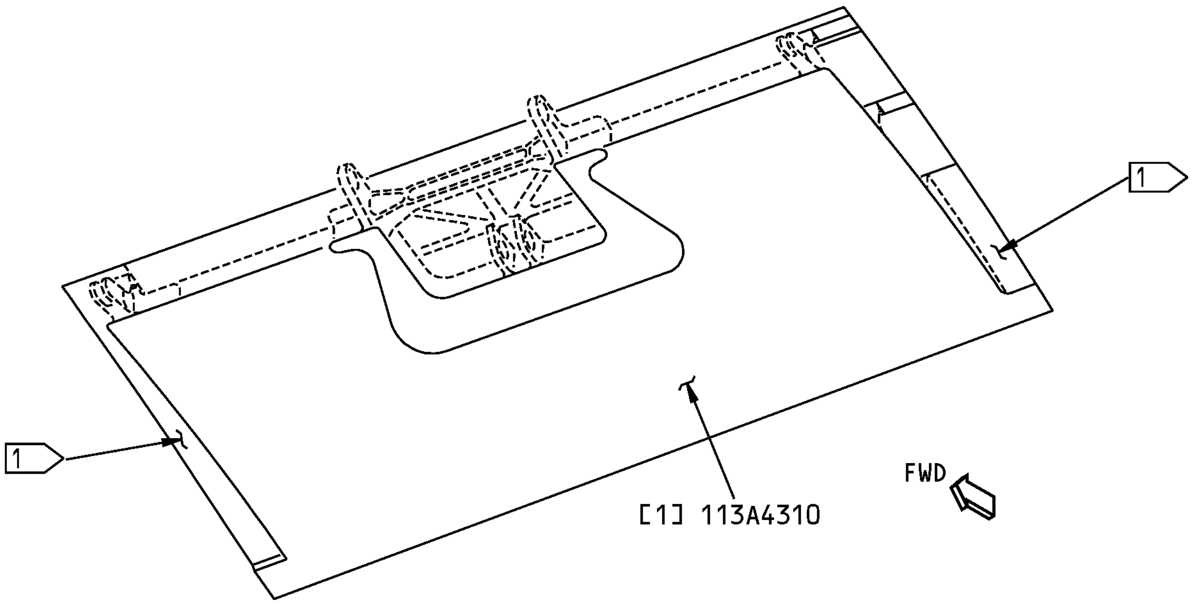
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

**Core Ribbon Direction for Figure 4, Item [1]**
**Figure 5**




NOTES

- REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERNCE)

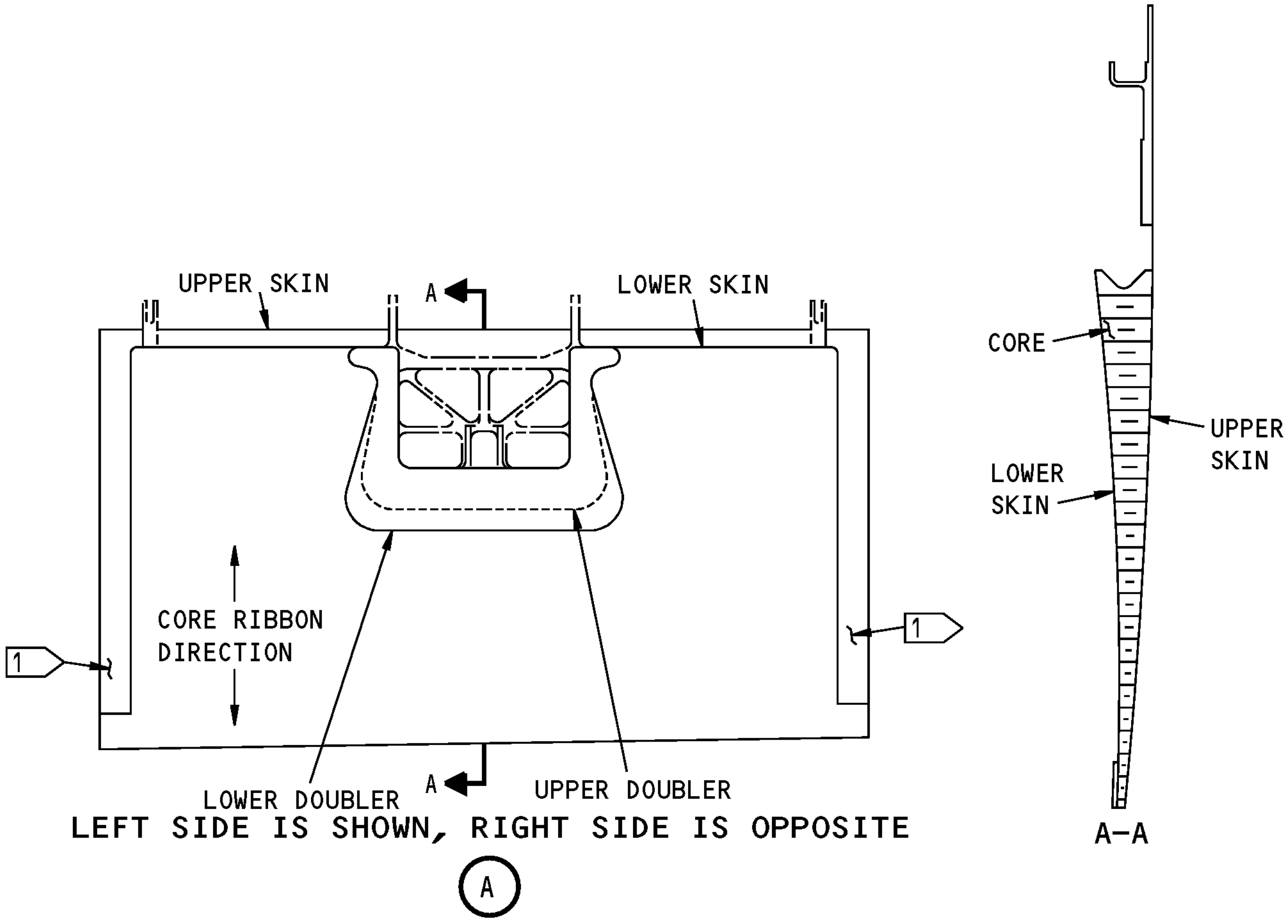
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outboard Spoiler Number 3 and 10 Skin Identification**
**Figure 6**

**Table 4**:

| LIST OF MATERIALS FOR FIGURE 6 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Spoiler Assembly, Number 3 and 10 Bonded Panel | | | |
| | Doubler - Lower | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Doubler - Upper | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - Lower | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - Upper | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 7 for the core ribbon direction | |

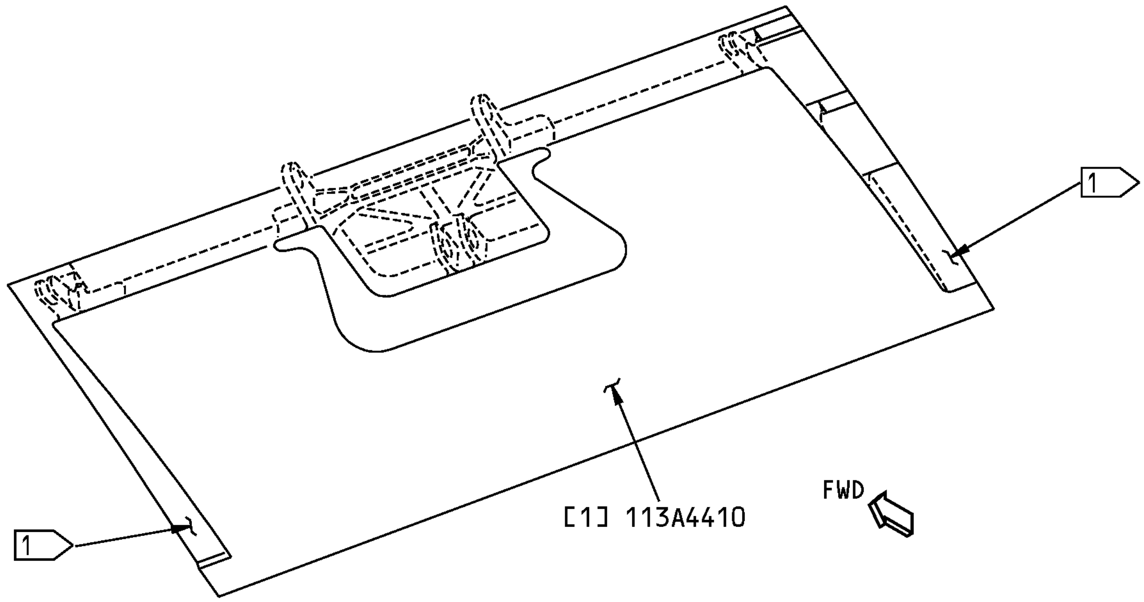
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

**Core Ribbon Direction for Figure 6, Item [1]**
**Figure 7**




NOTES

- REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

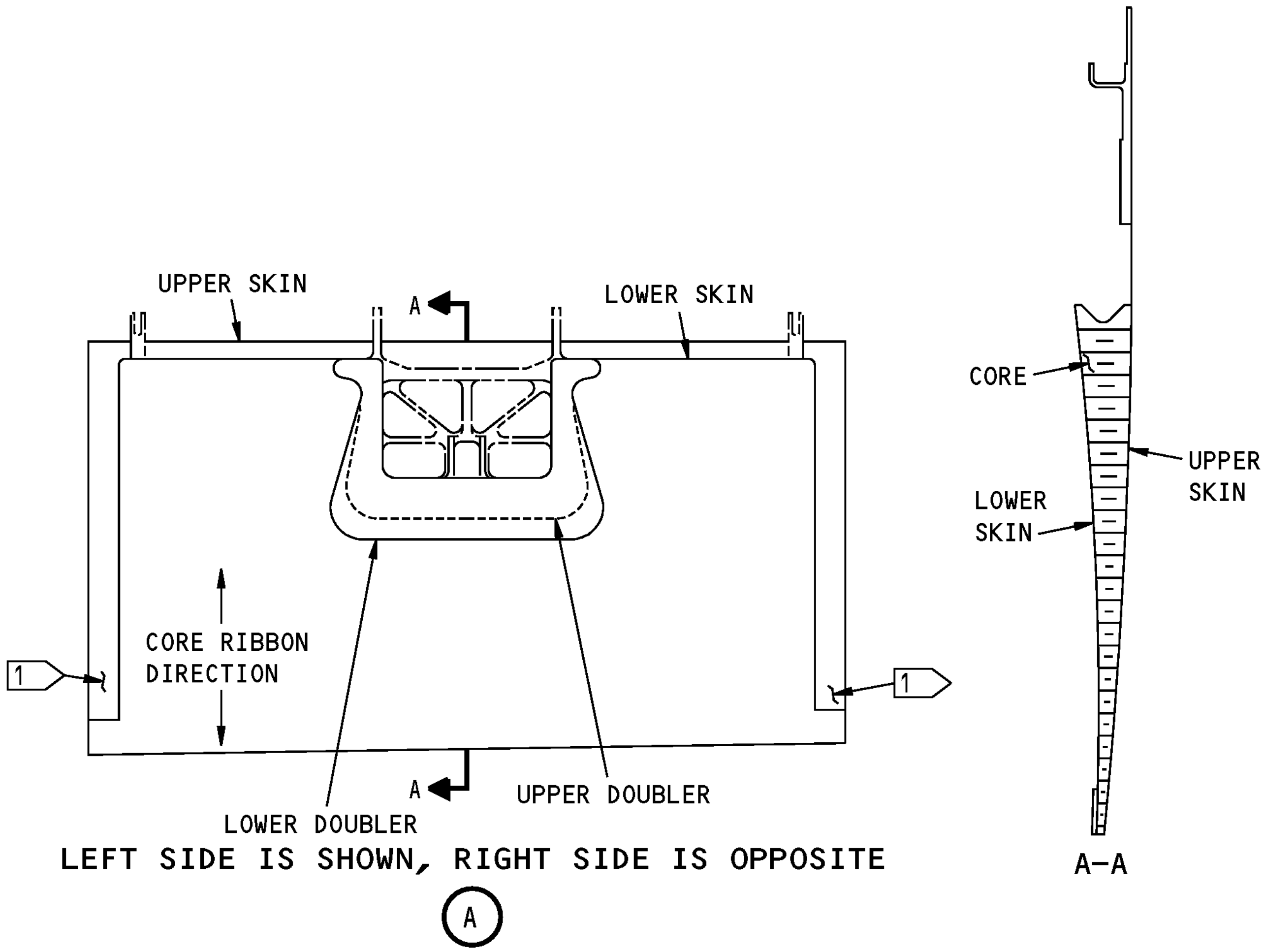
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outboard Spoiler Number 4 and 9 Skin Identification**
**Figure 8**

**Table 5**:

| LIST OF MATERIALS FOR FIGURE 8 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Spoiler Assembly, Number 4 and 9 Bonded Panel | | | |
| | Doubler - lower | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Doubler - Upper | 0.020 (0.058) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - lower | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - Upper | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 9 for the core ribbon direction | |

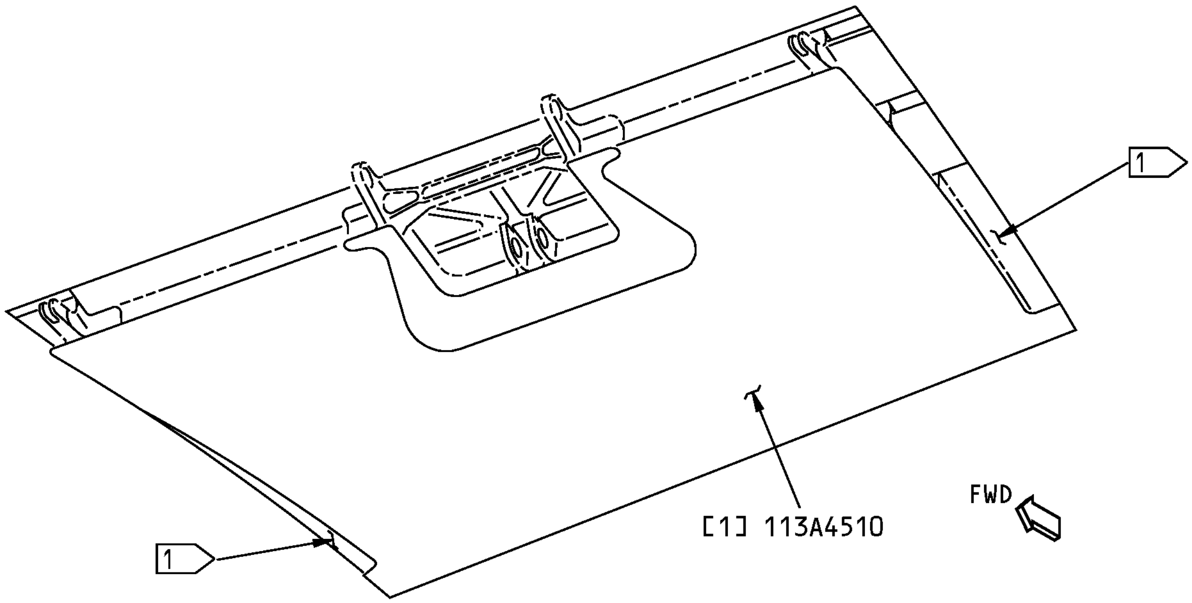
*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

**Core Ribbon Direction for Figure 8, Item [1]**
**Figure 9**




NOTES

- REFER TO TABLE 6 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

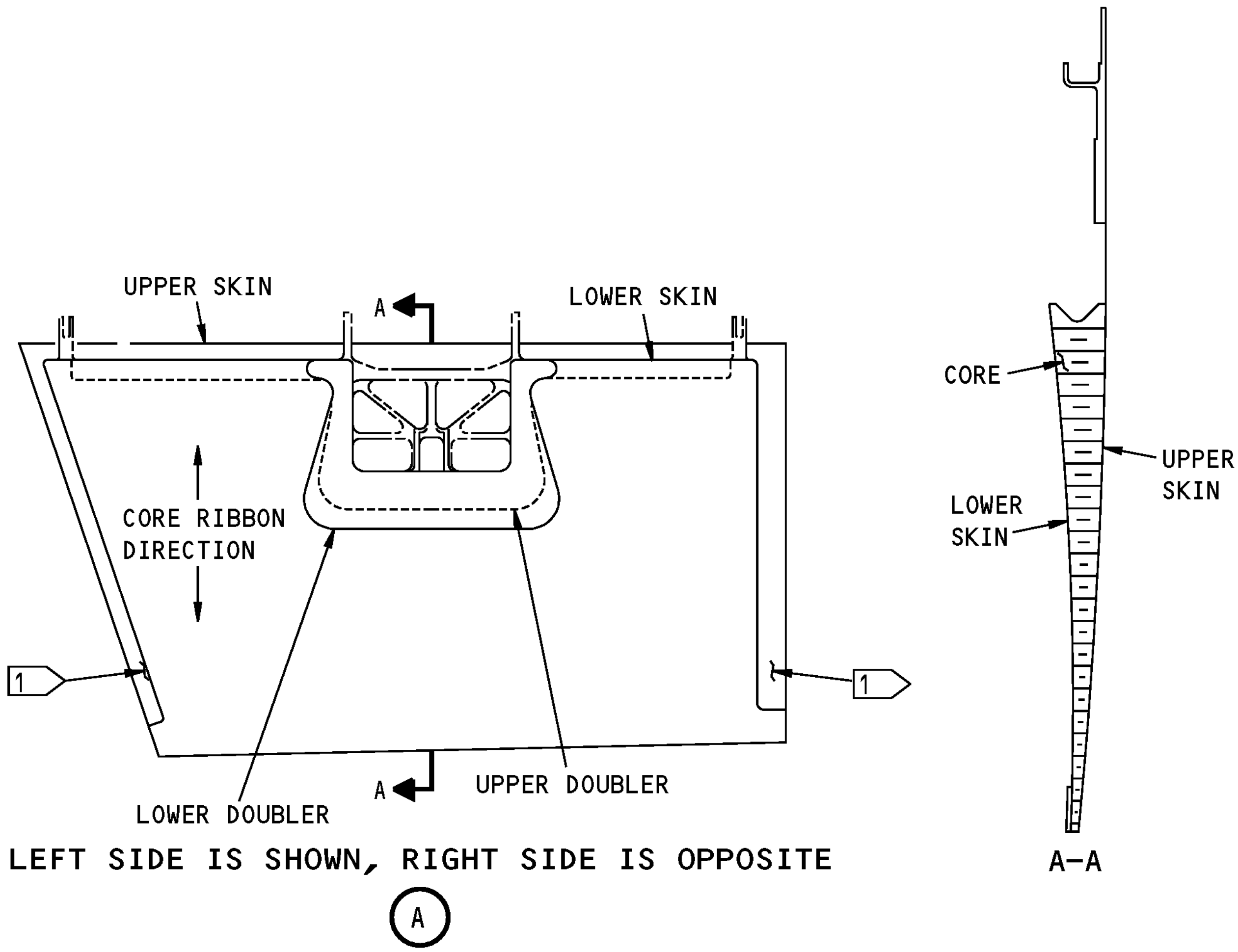
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outboard Spoiler Number 5 and 8 Skin Identification**
**Figure 10**

**Table 6**:

| LIST OF MATERIALS FOR FIGURE 10 | | | | |
|---|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **T*[1]** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Spoiler Assembly, Number 5 and 8 Bonded Panel | | | |
| | Doubler - lower | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Doubler - Upper | 0.020 (0.508) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - lower | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Skin - Upper | 0.016 (0.406) | 2024-T3 sheet as given in QQ-A-250/4 | |
| | Core | | 5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 11 for the core ribbon direction. | |

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).




NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

**Core Ribbon Direction for Figure 10, Item [1]**
**Figure 11**

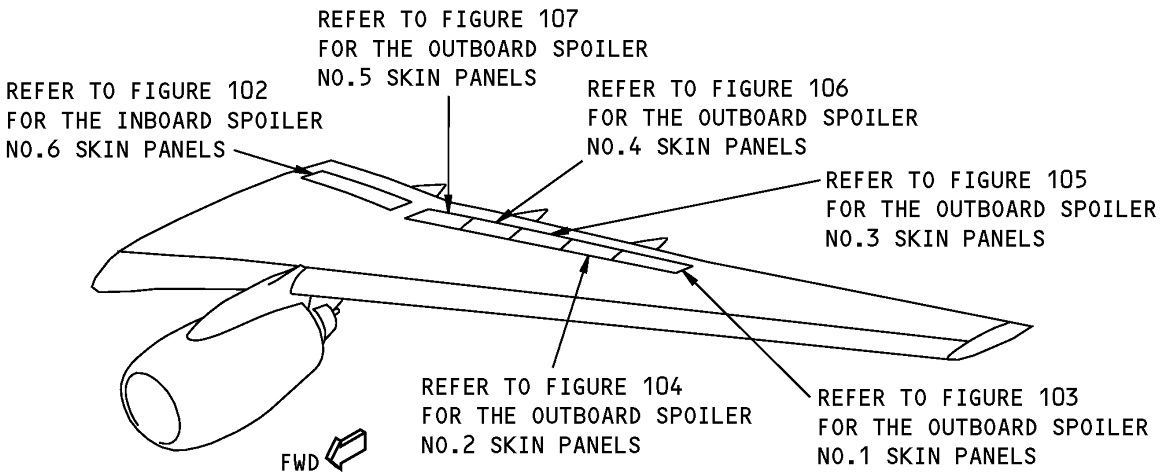
## ALLOWABLE DAMAGE 1 - INBOARD AND OUTBOARD SPOILER SKINS

### 1. Applicability

A. This subject gives the allowable damage limits for the inboard and outboard spoiler skins shown in Inboard and Outboard Spoiler Skin Allowable Damage, Figure 101/ALLOWABLE DAMAGE 1 and given in Table 101/ALLOWABLE DAMAGE 1.

**Table 101**:

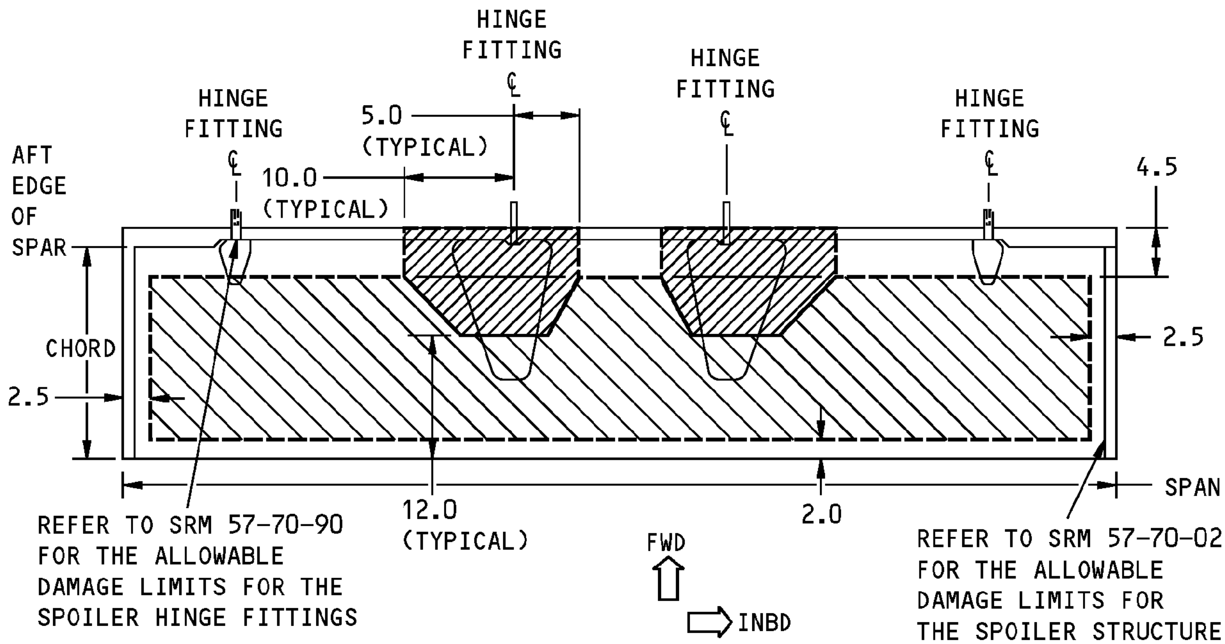
| PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS | | |
|---|---|---|
| TYPE OF STRUCTURE | ZONE LOCATION | PARAGRAPH |
| ALUMINUM SKIN AREAS | 1 | 4.A |
| | 2 | 4.B |
| | 3 | 4.C |



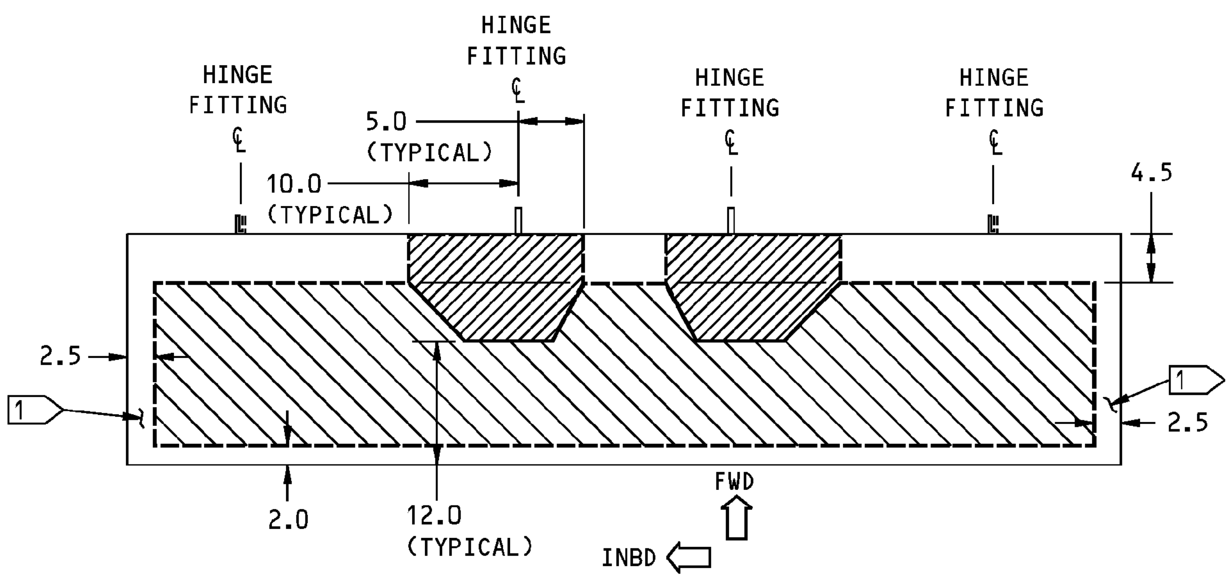

NOTE: REFER TO TABLE 101 FOR THE ALLOWABLE DAMAGE LIMITS PARAGRAPH THAT IS APPLICABLE TO EACH ZONE.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Inboard and Outboard Spoiler Skin Allowable Damage**
**Figure 101**




LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

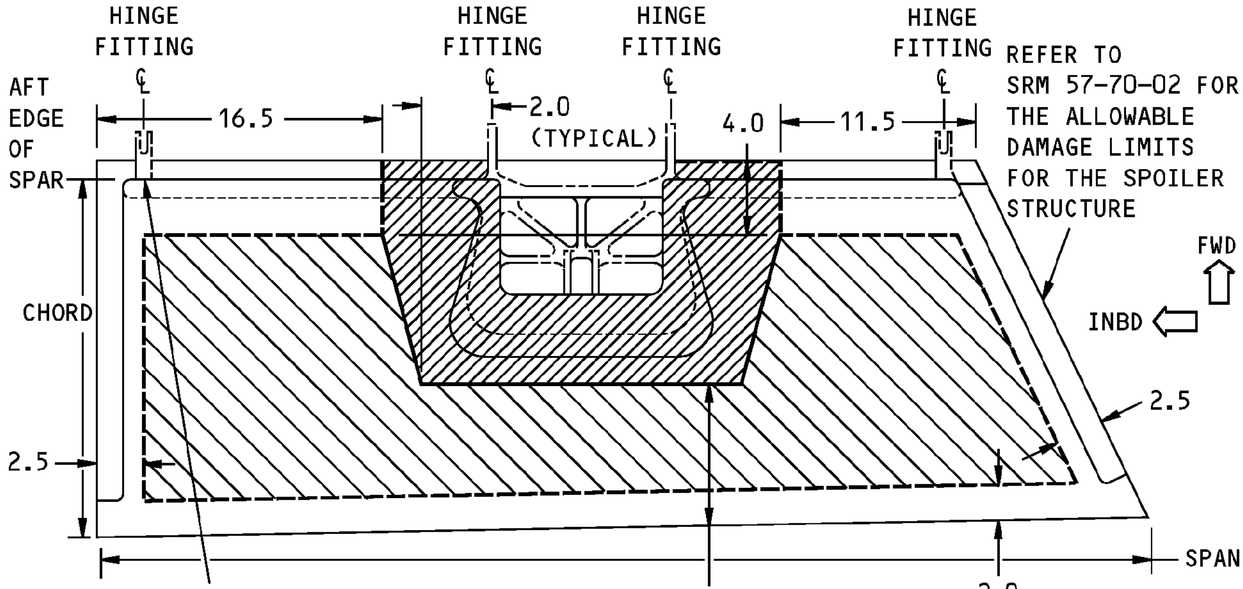



LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

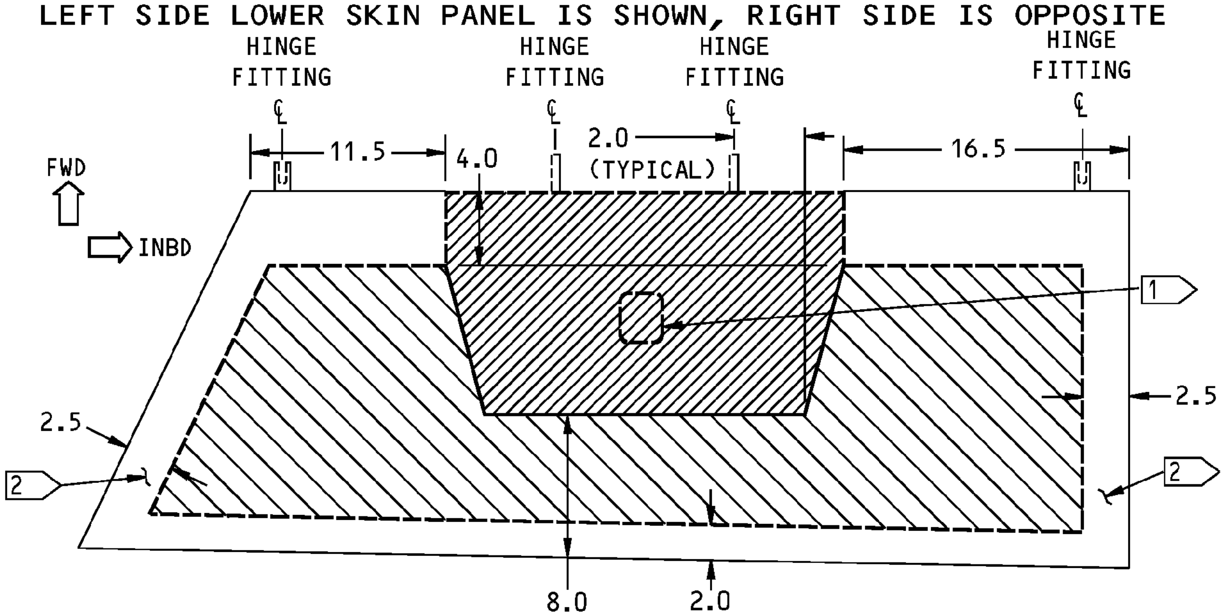
ZONE 1 ZONE 2 ZONE 3

NOTES
- ALL DIMENSIONS ARE IN INCHES
- [1] 0.005 INCH DEPRESSION (REFERENCE)

**Inboard Spoiler No.6 and 7 Skin Allowable Damage**
**Figure 102**




LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE




LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

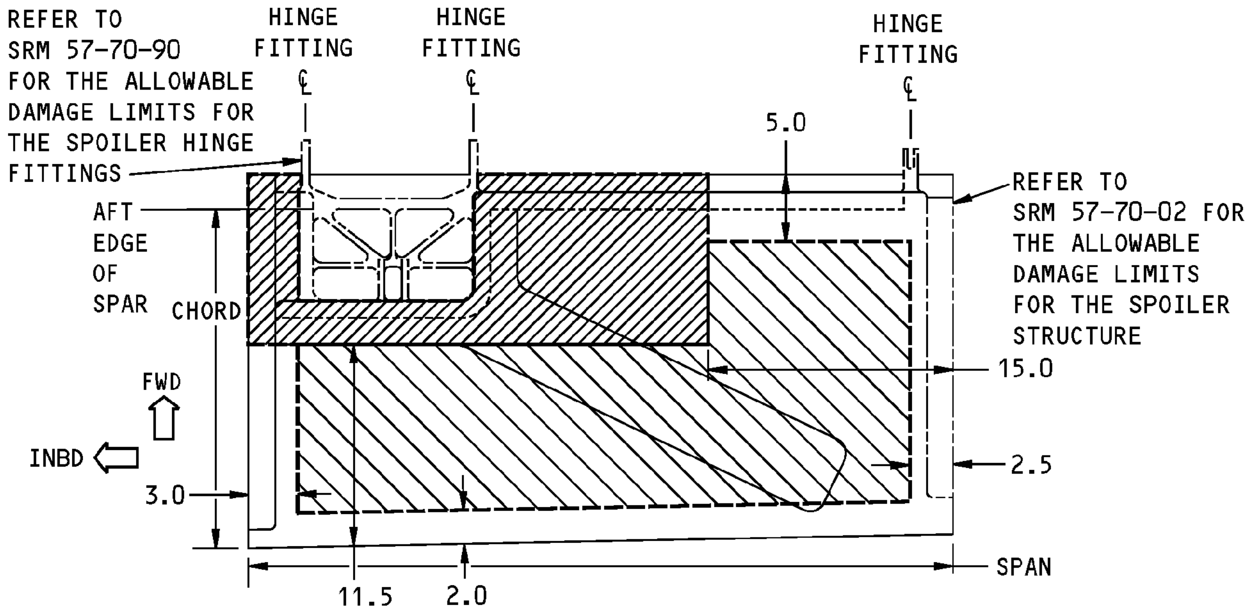
NOTES

- ALL DIMENSIONS ARE IN INCHES

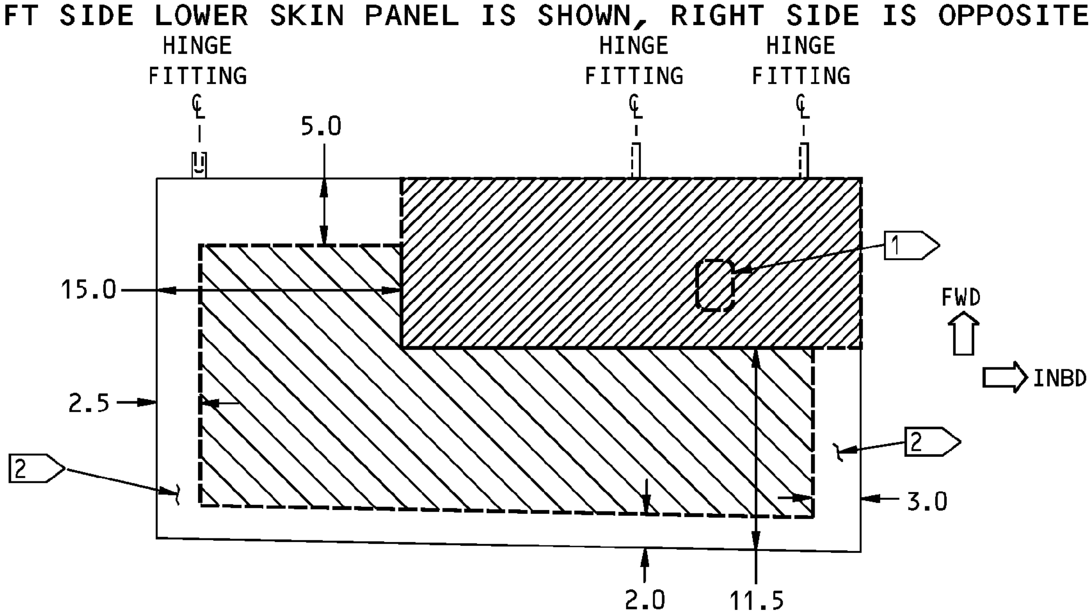
1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

**Outboard Spoiler No.1 and 12 Skin Allowable Damage**
**Figure 103**




LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE




LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

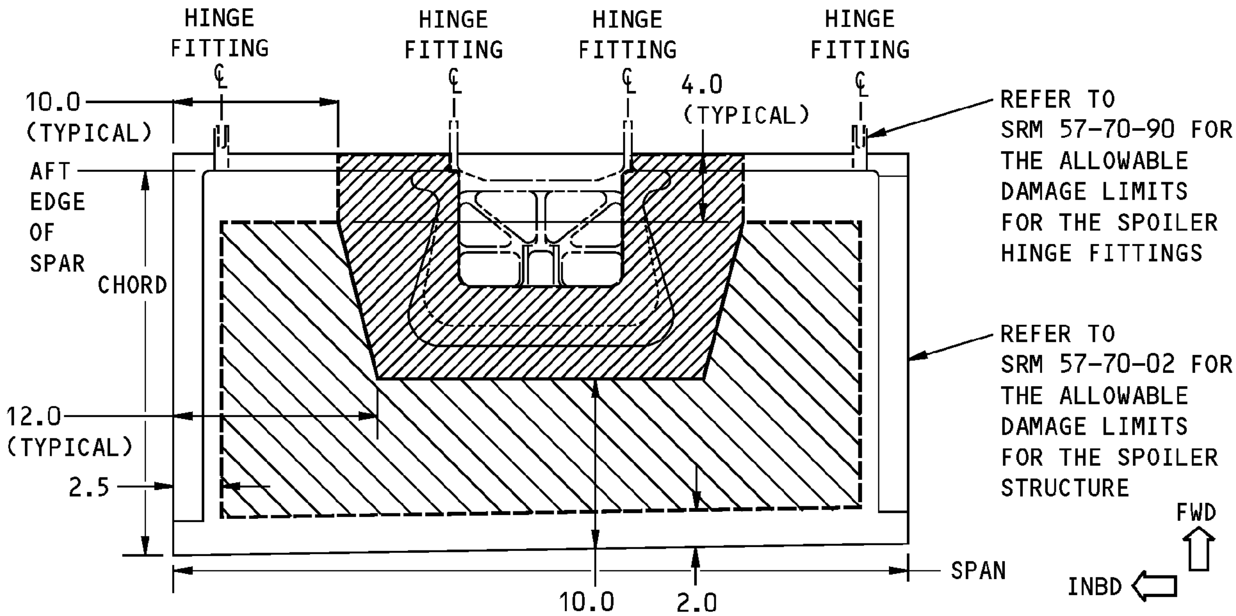
NOTES

- ALL DIMENSIONS ARE IN INCHES

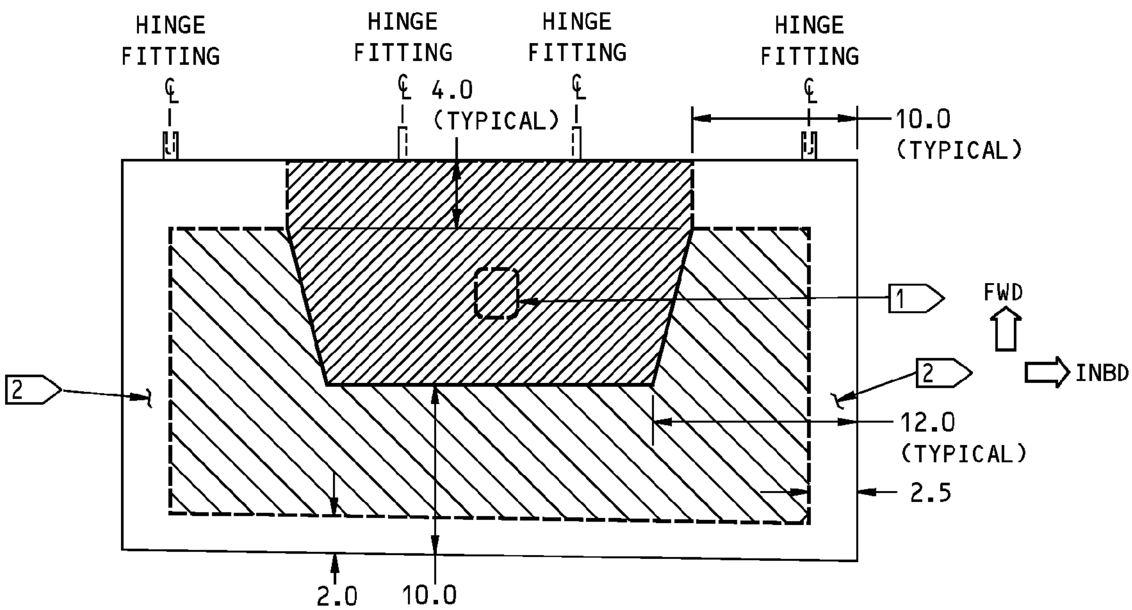
1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

**Outboard Spoiler No.2 and 11 Skin Allowable Damage**
**Figure 104**




LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE




LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

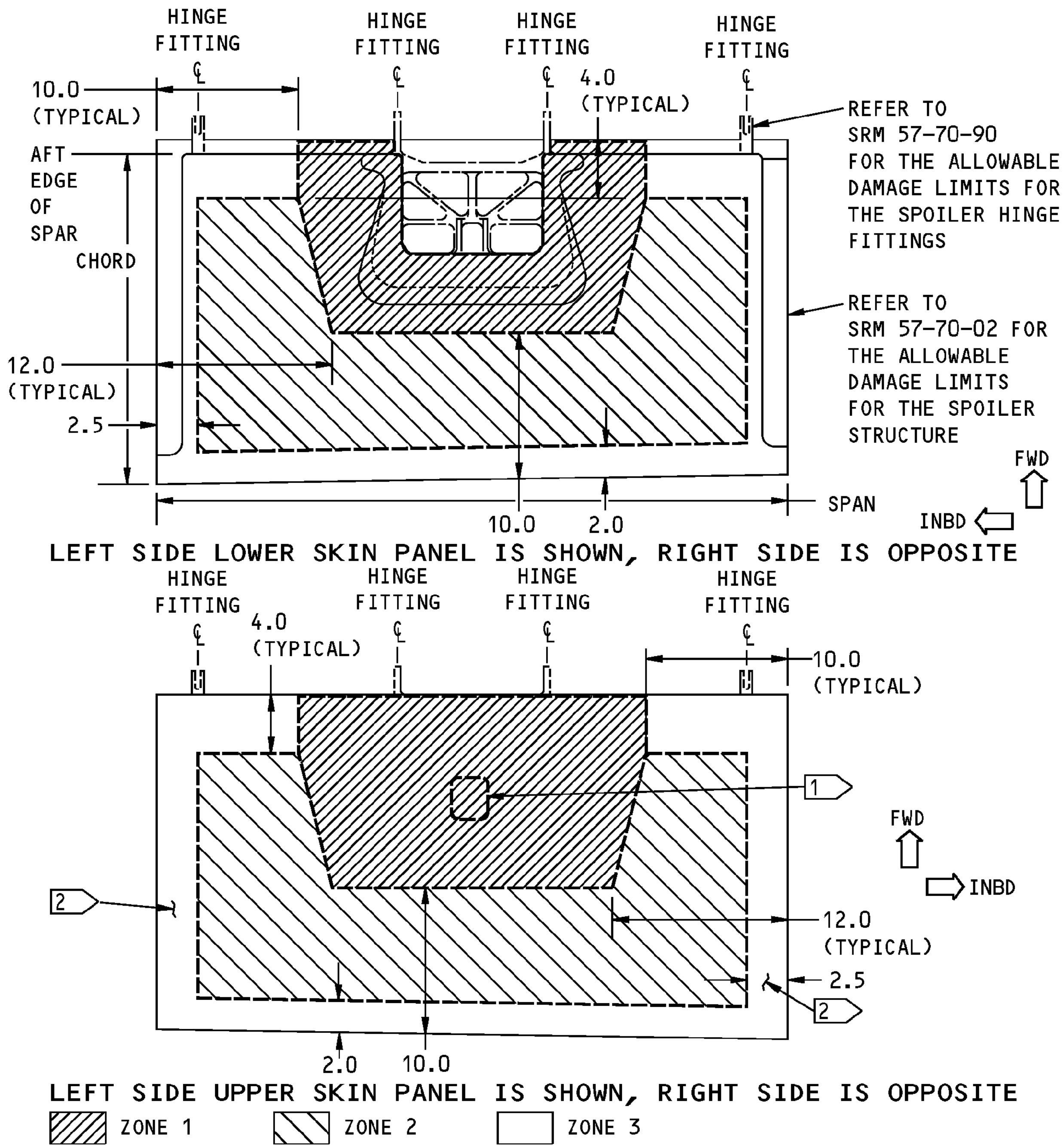
NOTES

- ALL DIMENSIONS ARE IN INCHES

1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

**Outboard Spoiler No.3 and 10 Skin Allowable Damage**
**Figure 105**



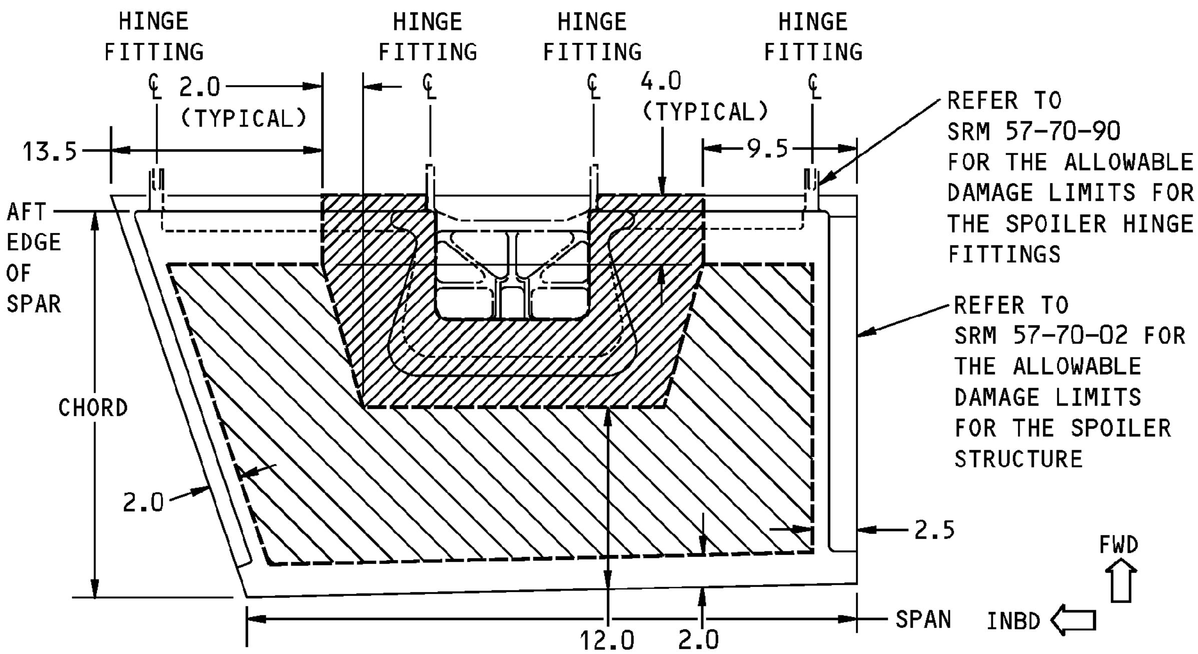

NOTES

- ALL DIMENSIONS ARE IN INCHES

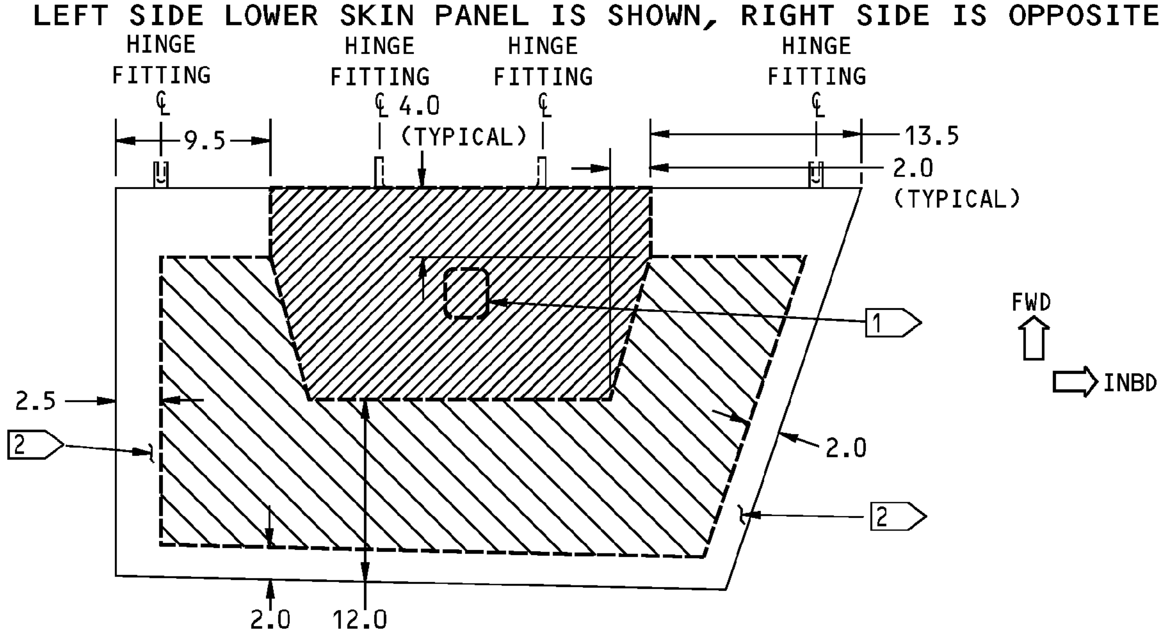
1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

**Outboard Spoiler No.4 and 9 Skin Allowable Damage**
**Figure 106**




LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE




LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1    ZONE 2    ZONE 3

NOTES

- ALL DIMENSIONS ARE IN INCHES

1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

**Outboard Spoiler No.5 and 8 Skin Allowable Damage**
**Figure 107**

## 2. General

A. Do the steps that follow if you have damage to the aluminum skins.

(1) Remove the damaged material.

(a) Refer to 51-10-02 for the procedures.

(b) Refer to 51-30-03 for the sources of nonmetallic materials you need to remove the damage.

(c) Refer to 51-30-05 for the sources of the equipment and tools you need to remove the damage.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply one layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

B. Do the steps that follow if you remove damage from the skin and honeycomb core:

(1) Remove 0.25 inch of the honeycomb core at the edge of the cutout to make a recess.

(2) Fill the recess with BMS 5-28, Type 15 or 17 potting compound. Refer to 51-70-10, Repair General.

(3) As an alternative, fill the recess with BMS 5-95 sealant. Refer to 51-20-05.

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| 51-70-01 | REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS |
| 51-70-10 | ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Zone 1 - Skin Panel Surface

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits for Zone 1, Figure 108/ALLOWABLE DAMAGE 1, Detail A if the largest dimension of the damage is not more than 3.0 inches.

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits for Zone 1, Figure 108/ALLOWABLE DAMAGE 1, Detail B if:

1) The depth of the damage is a maximum of 0.10 inch

2) The maximum diameter of the damage is 0.50 inch

3) The damage area is not at an attachment to the spoiler structure.

(b) If the damage depth is more than 0.10 inch and less than 0.25 inch, do the steps that follow:

1) Fill or rework the dent as given in 51-70-01.

2) If you fill the dent, do the steps that follow:

a) Seal the damage with aluminum foil tape (speed tape 3M-436 or the equivalent).

b) Make an inspection of the dent at each 400 flight hour interval, or more frequently.

- Make sure the tape is in satisfactory condition.
- Repair the dent if the damage becomes larger.
- If the damage has not become larger, repair the damage in 5000 flight hours or less.

(4) Holes and Punctures are not permitted.

(5) Delamination is not permitted.

B. Zone 2 - Skin Panel Surface

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits for Zone 2, Figure 109/ALLOWABLE DAMAGE 1, Detail A if the condition that follows is true:

1) The largest dimension of the damage is not more than 25 percent of the length of the panel chord.

(b) Scratches are permitted if the conditions that follow are true:

1) Two or more scratches do not go across each other

2) The damage is not less than 1.50 inches away from:

a) A material edge

b) A potted area

c) An attachment to the spoiler structure.

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits for Zone 2, Figure 109/ALLOWABLE DAMAGE 1, Detail B if:

1) The depth of damage is a maximum of 0.10 inch

2) The maximum diameter of the damage is 25 percent of the panel chord

3) The distance between two dents, or between a dent and other damage is a minimum of 0.5D (D = the large dimension of the two damage areas)

4) The damage area is not at an attachment to the spoiler structure.

(b) If the damage depth is more than 0.10 inch and less than 0.25 inch, do the steps that follow:

1) Fill or rework the dent as given in 51-70-01.

2) If you fill the dent, do the steps that follow:

a) Seal the damage with aluminum foil tape (speed tape 3M-436 or the equivalent).

b) Make an inspection of the dent at each 400 flight hour interval, or more frequently.

- Make sure the tape is in satisfactory condition.
- Repair the dent if the damage becomes larger.
- Repair the damage in 5000 flight hours or less.

(4) Holes and Punctures:

(a) The damage is permitted up to a maximum length of 15 percent of the length of the panel chord if you do as follows:

1) Trim the skin around the damage to a smooth circular or oval shape.

2) Clean and remove the water from the damaged area.

3) Apply aluminum foil tape (speed tape 3M-436 or the equivalent).

a) Keep a record of the damage location.

b) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 5000 flight hours or less.

(5) Delamination:

(a) The damage is permitted as shown in Allowable Damage Limits for Zone 2, Figure 109/ALLOWABLE DAMAGE 1, Detail C if you do as follows:

1) Make an inspection of the damage at each 400 flight hour interval, or more frequently.

a) Repair the damage in 5000 flight hours or less.

C. Zone 3 - Skin Panel Edges

(1) Cracks:

(a) For panel edges other than the trailing edge:

1) Remove the damage as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Details A and B.

2) Cracks are not permitted aft of the spar.

(b) For the trailing edge of the panel:

1) The length of the damage must not be more than 2.0 inches.

2) The damage must not be less than 5D (D = largest damage dimension) from other damage.

3) Remove water from the damage area and seal as follows:

a) Apply one layer of speed tape (3M-436 or the equivalent) that extends 2.0 inches all around the damage.

b) Apply a second layer of speed tape that extends 1.0 inch all around the first layer of speed tape.

4) Make an inspection of and repair the damage.

a) Keep a record of the damage location.

b) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

c) Repair the damage in 5000 flight hours or less.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Details A and B.

1) The damage must not be less than 1.50 inches from a fastener hole.

(3) Dents:

(a) For panel edges other than the trailing edges, the damage is permitted if:

1) The largest dimension of damage is 15 percent of the length of the panel chord

2) The largest dimension of damage is 0.50 inch in depth

3) The distance between two dents, or a dent and other damage, is not less than 0.5D (D = The larger dimension of the two damage areas)

**NOTE**: If delamination is found at a dent location, use the limits for delamination.

4) You clean and remove the water from the damage area

5) You apply aluminum foil tape (speed tape 3M-436 or the equivalent).

a) Keep a record of the damage location.

b) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 5000 flight hours or less if the damage has not become larger.

(b) For the trailing edge of the panels, the damage is permitted if:

1) The largest dimension of damage is 15 percent of the length of the panel span.

2) The largest dimension of damage is 0.50 inch in depth.

3) The distance between two dents, or a dent and other damage, is not less than 0.5D (D = The larger dimension of the two damage areas)

**NOTE**: If delamination is found at a dent location, use the limits for delamination.

4) You clean and remove the water from the damaged area

5) You apply aluminum foil tape (speed tape 3M-436 or the equivalent).

a) Keep a record of the damage location.

b) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 5000 flight hours or less if the damage has not become larger.

(4) Holes and Punctures:

(a) The damage is permitted as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail C.

(b) Apply aluminum foil tape (speed tape 3M-436 or the equivalent).

1) Keep a record of the damage location.

2) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

3) Repair the damage in 5 flight hours or less if the damage becomes larger.

4) Replace the tape as necessary.

5) Repair the damage in 5000 flight hours or less if the damage has not become larger.

(5) Delamination:

(a) Remove the damage as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail D with the limits that follow:

1) The length of the damage must not be more than 2.0 inches.

2) The width of the damage must not be more than 1.50 inches.

3) The damage is not less than 5D (D = largest damage dimension) from other damage.

(b) Seal the damage as follows:

**NOTE**: Refer to Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail D for the definition of the damage dimensions.

1) Apply one layer of speed tape (3M-436 or the equivalent) that extends 2.0 inches all around the damage.

2) Apply a second layer of speed tape that extends 1.0 inch all around the first layer of speed tape.

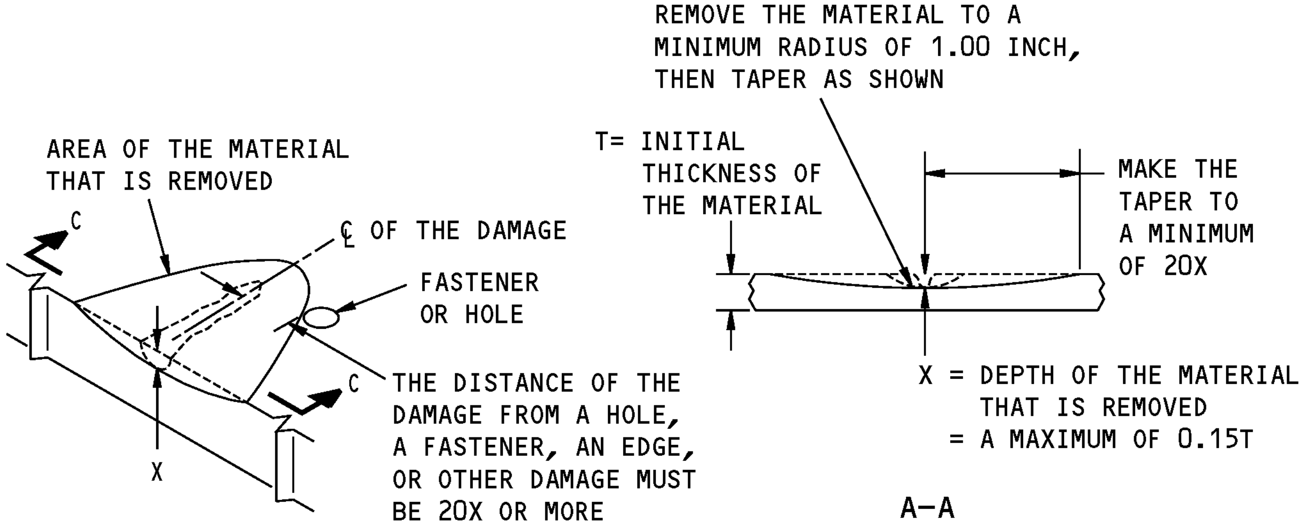
(c) Make an inspection of and repair the damage.

1) Keep a record of the damage location.

2) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

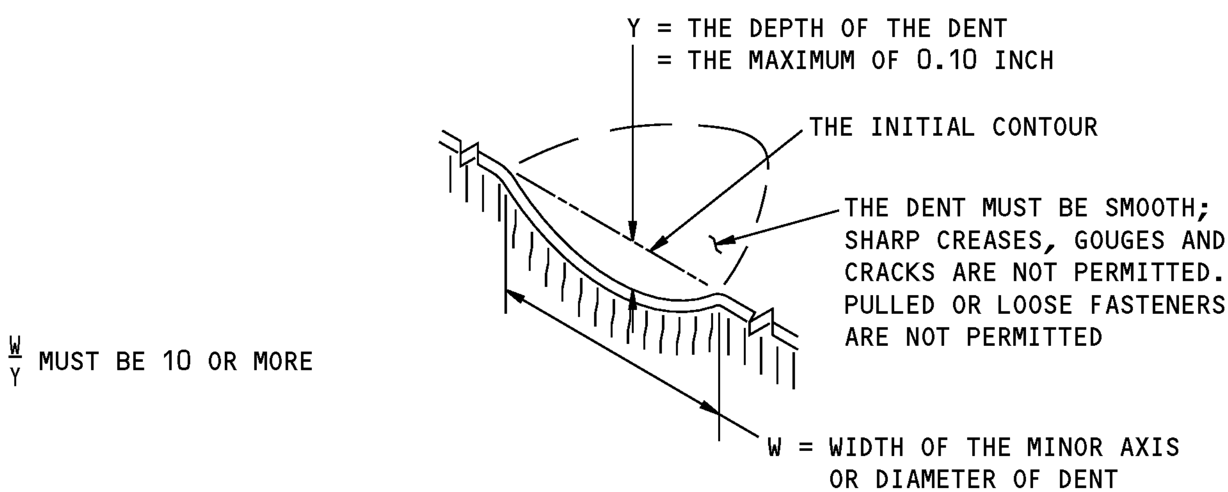
3) Repair the damage if the damage becomes larger.

4) Repair the damage in 5000 flight hours or less if the damage has not become larger.




REMOVAL OF DAMAGED MATERIAL ON A SURFACE

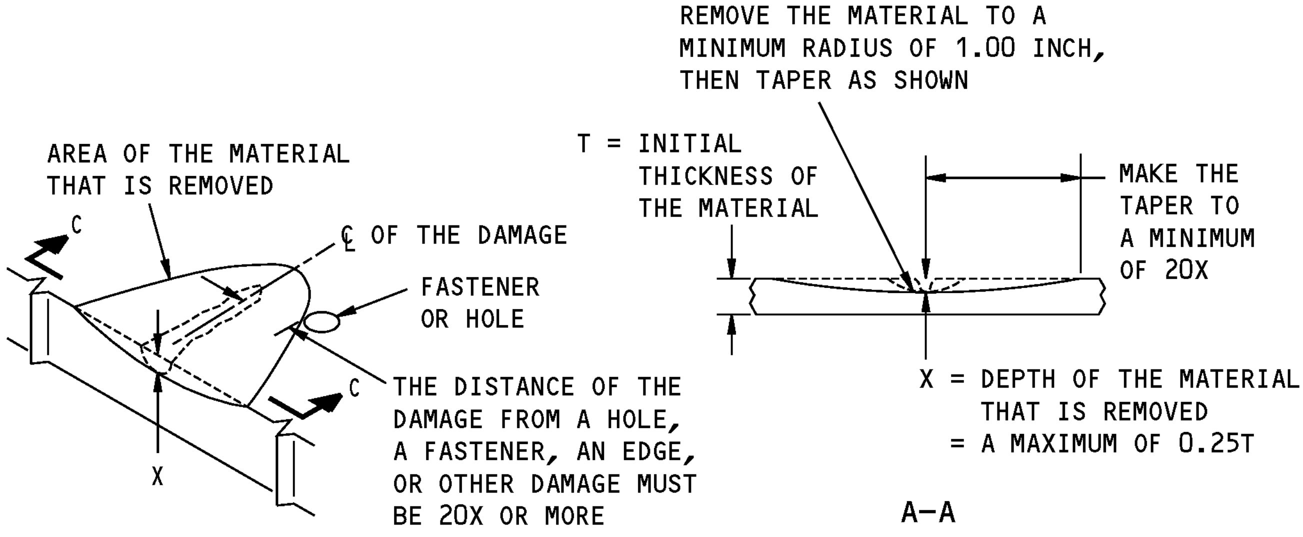
A




DENT THAT IS PERMITTED

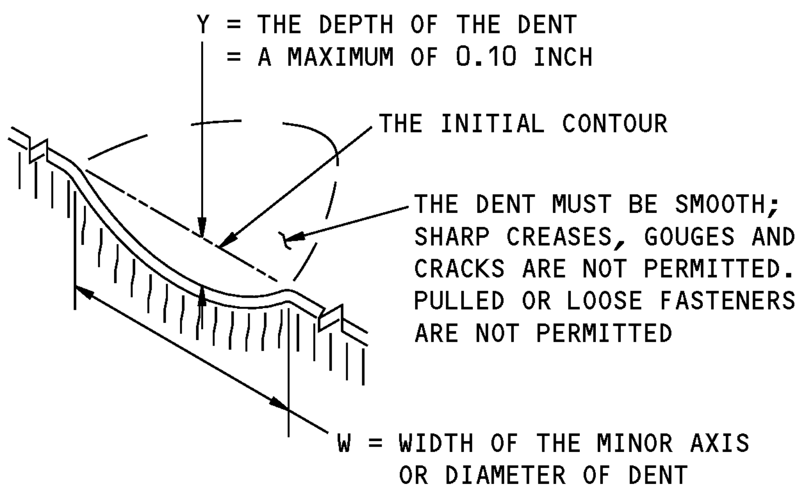
B

**Allowable Damage Limits for Zone 1**
**Figure 108**




REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

A

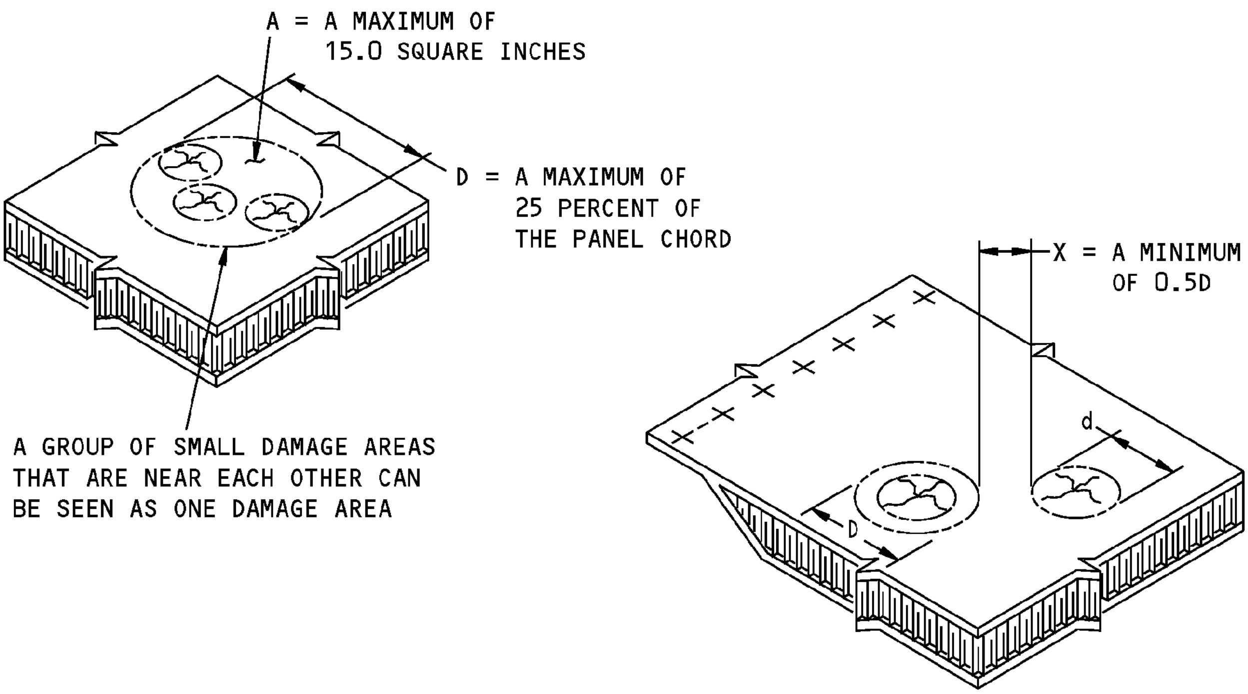



$\frac{W}{Y}$ MUST BE 10 OR MORE

DENT THAT IS PERMITTED

B

**Allowable Damage Limits for Zone 2**
**Figure 109 (Sheet 1 of 2)**




NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

A = THE AREA OF THE DAMAGE.

D = THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

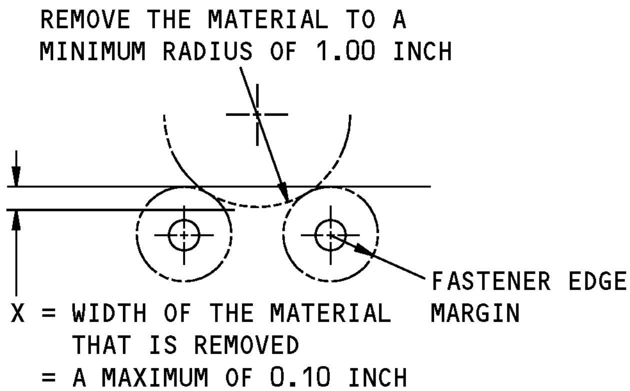
d = THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X = THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

DAMAGE THAT IS PERMITTED ON THE SURFACE OF
AN ALUMINUM HONEYCOMB PANEL

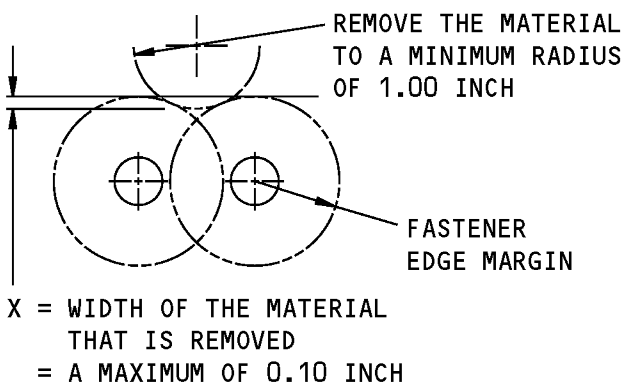
C

**Allowable Damage Limits for Zone 2**
**Figure 109 (Sheet 2 of 2)**




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

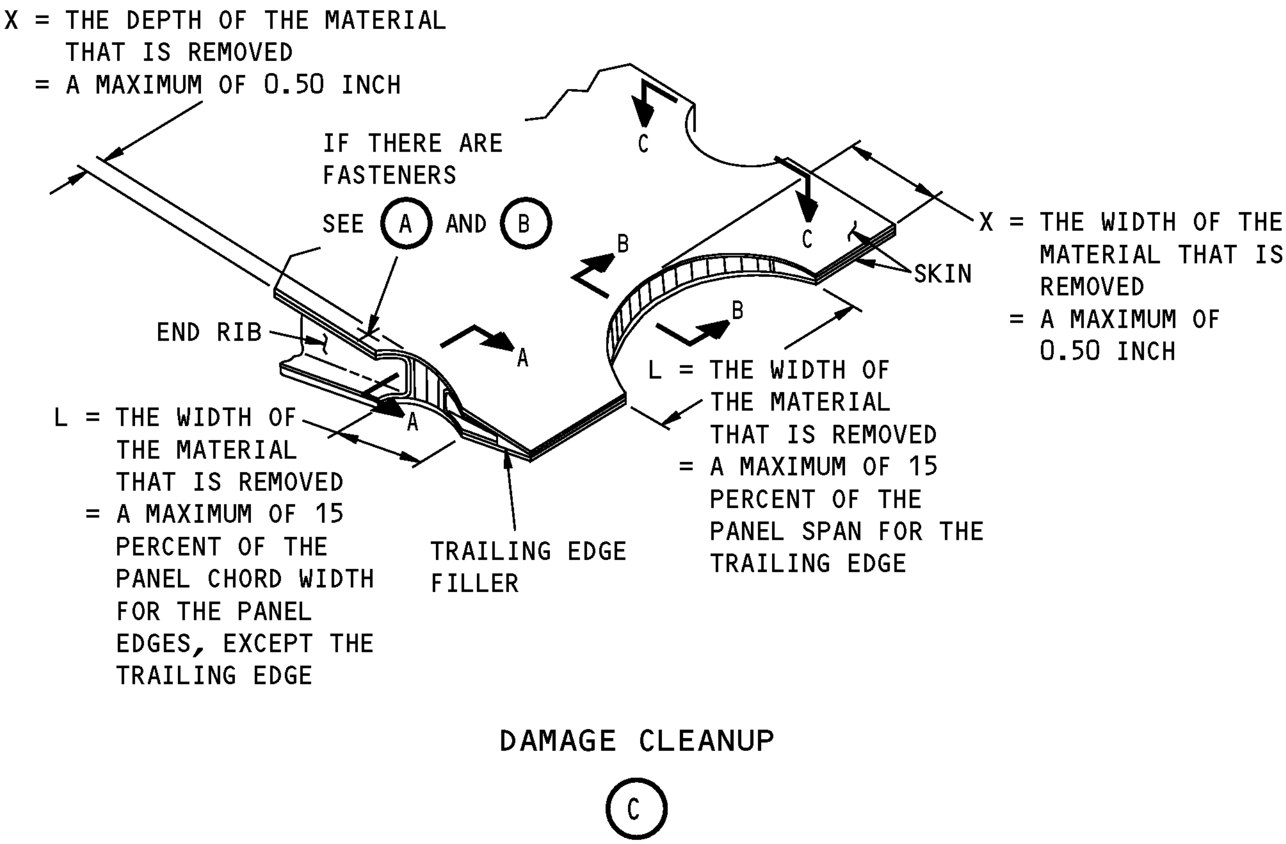
A

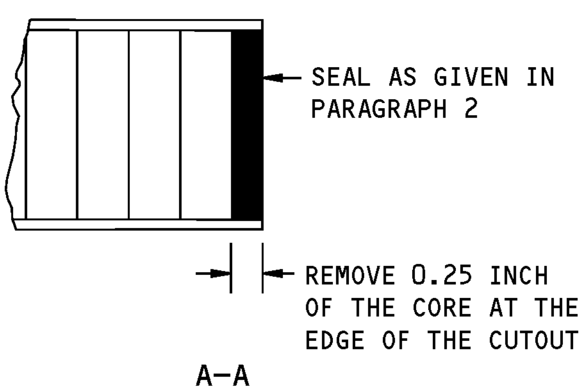


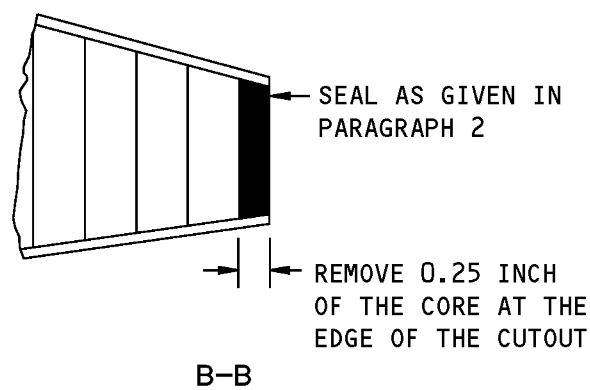

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

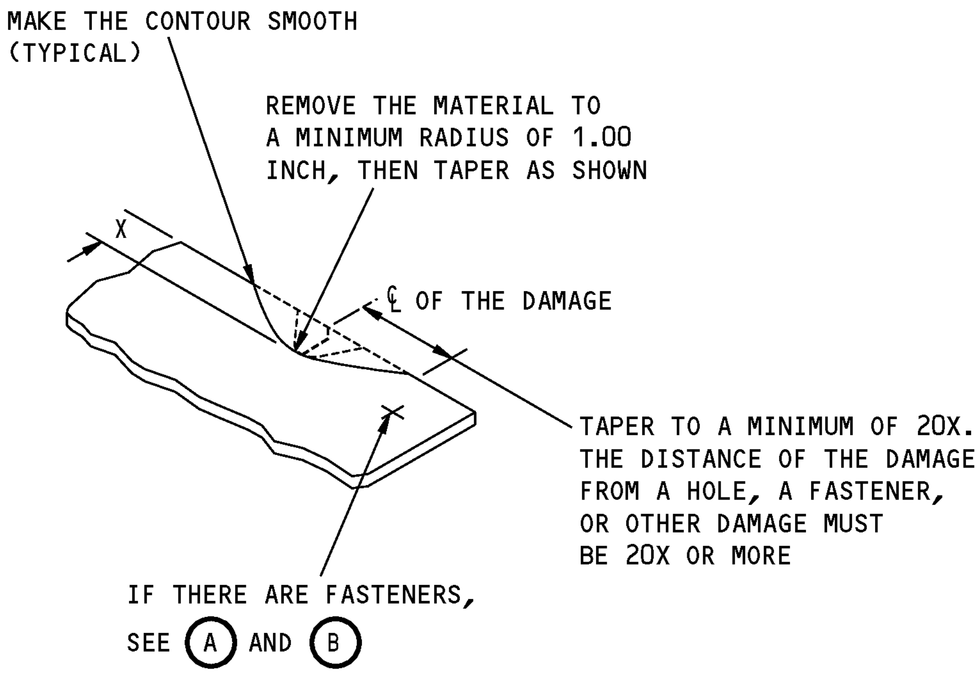
**Allowable Damage Limits for Zone 3**
**Figure 110 (Sheet 1 of 4)**




**Allowable Damage Limits for Zone 3**
**Figure 110 (Sheet 2 of 4)**

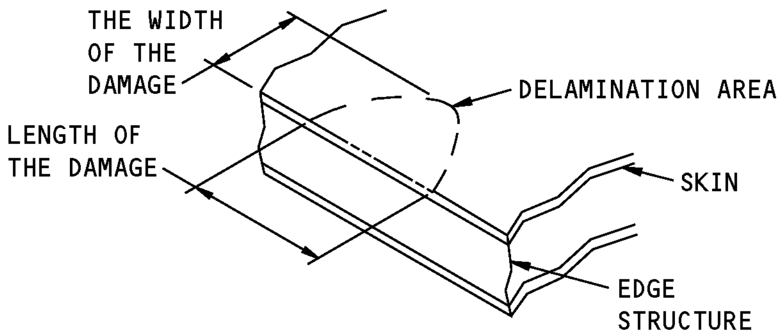



X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.50 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE ONLY
(NOT APPLICABLE IF HONEYCOMB CORE MATERIAL IS REMOVED)

C-C

**Allowable Damage Limits for Zone 3**
**Figure 110 (Sheet 3 of 4)**

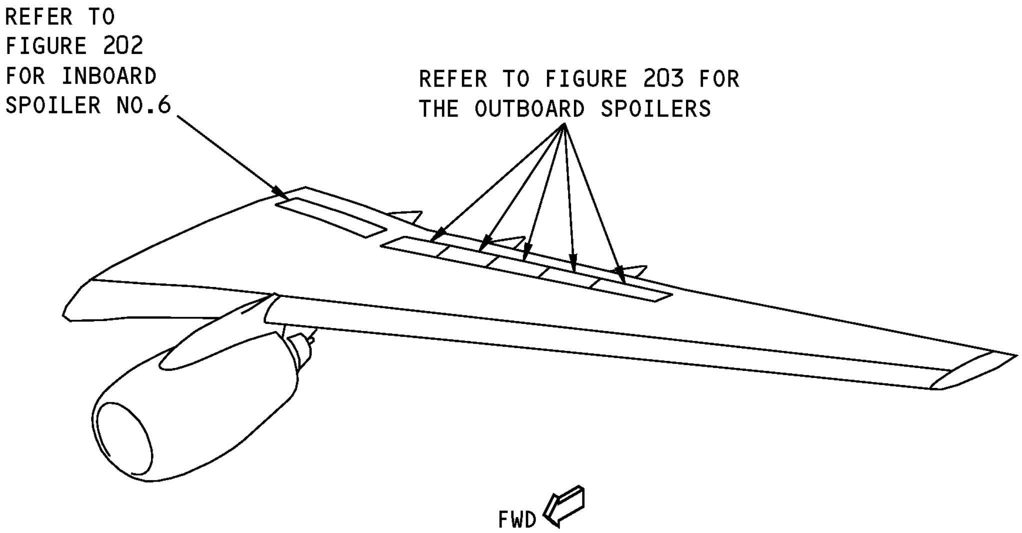



DEFINITIONS OF THE DAMAGE DIMENSIONS
FOR A DELAMINATION AT AN EDGE

D

**Allowable Damage Limits for Zone 3**
**Figure 110 (Sheet 4 of 4)**

## REPAIR 1 - INBOARD AND OUTBOARD SPOILER SKINS

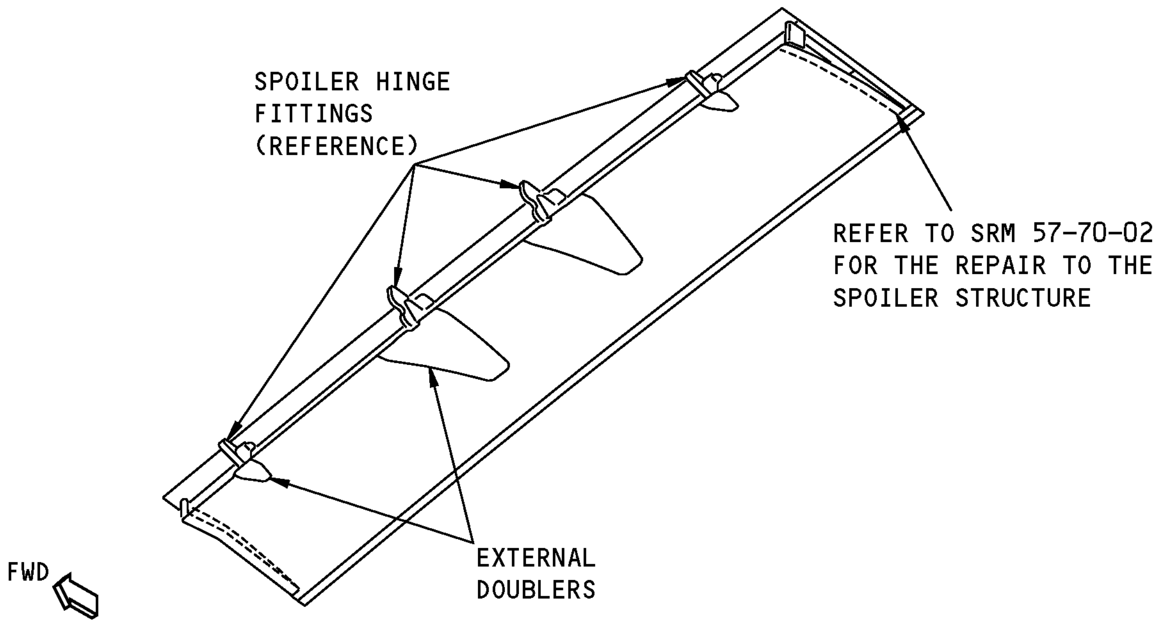



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTES

- REFER TO SRM 51-70-10 FOR TYPICAL METAL REPAIRS FOR THE SPOILER SKIN. MAKE SURE THERE IS SUFFICIENT CLEARANCE FROM ADJACENT STRUCTURE TO INSTALL THE REPAIR PARTS.
- THERE ARE NO REPAIRS IN THE STRUCTURE REPAIR MANUAL AT THIS TIME FOR TWO AREAS:
  - EXTERNAL DOUBLERS
  - AREAS THAT ARE 3.0 INCHES OR LESS FROM A FITTING

**Inboard and Outboard Spoiler Skin Locations**
**Figure 201**

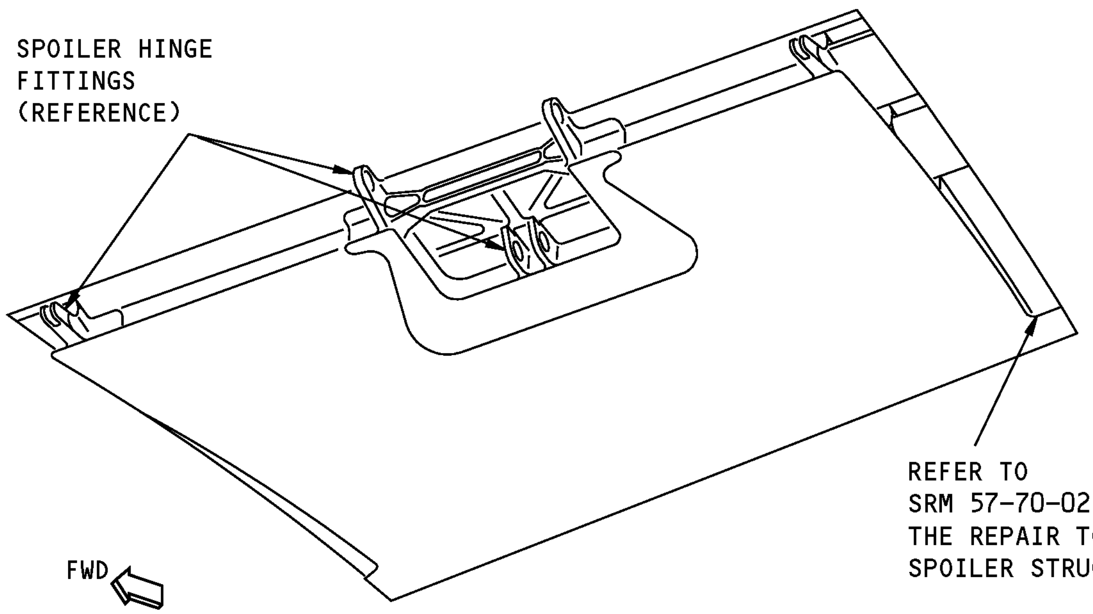



NOTE: REFER TO THE IDENTIFICATION SECTION FOR THE SHAPES OF THE SPOILER.

LOWER SURFACE OF THE LEFT SIDE IS SHOWN,

RIGHT SIDE IS OPPOSITE

**Inboard Spoiler No.6 Skin Location**
**Figure 202**

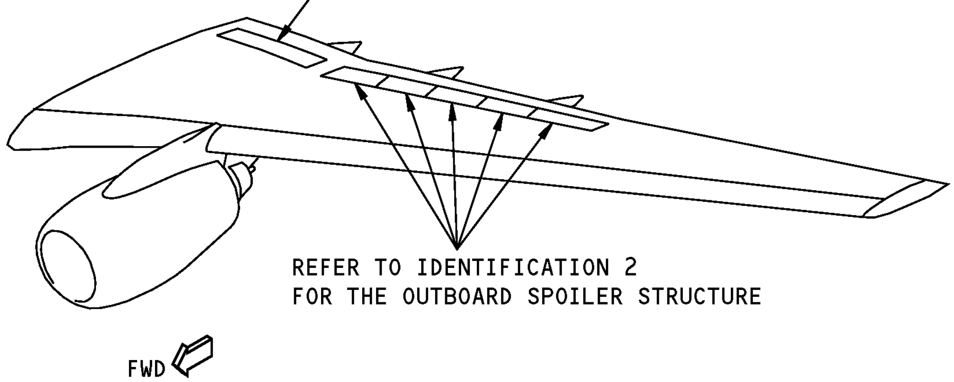



NOTE: REFER TO THE IDENTIFICATION SECTION FOR THE SHAPES OF THE SPOILER.

LOWER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE
A TYPICAL OUTBOARD SPOILER

**Outboard Spoiler Skin Location**
**Figure 203**

## IDENTIFICATION 1 - INBOARD SPOILER STRUCTURE



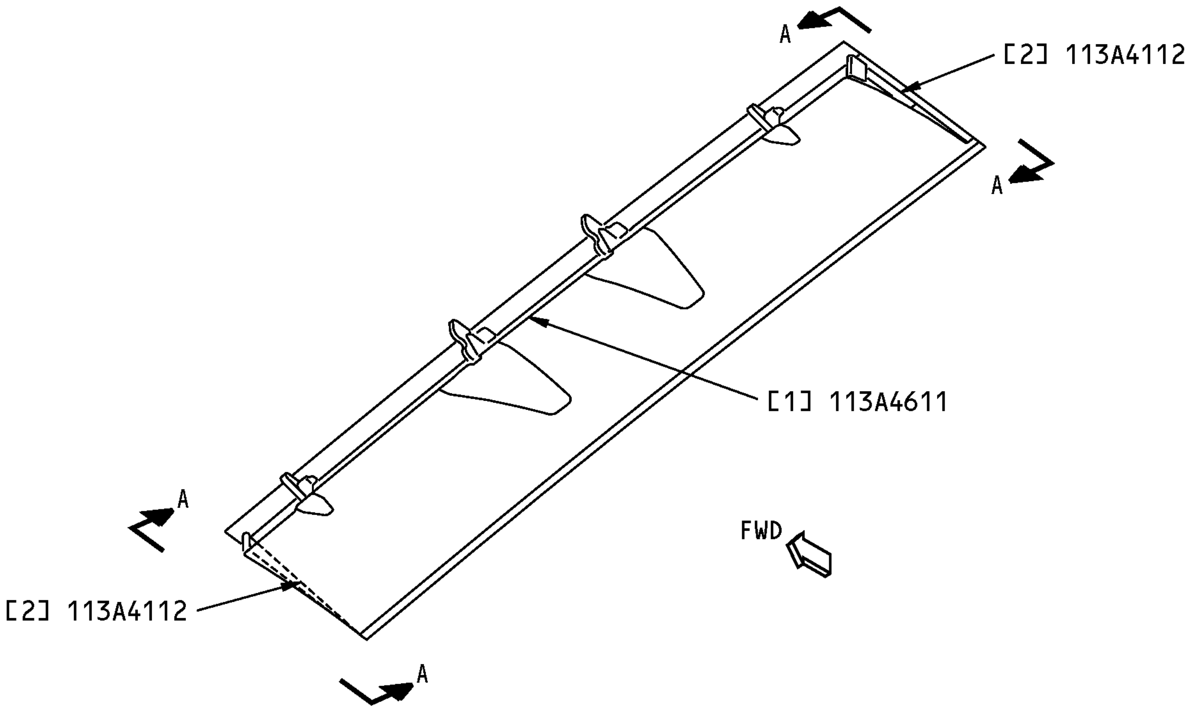

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

**Inboard Spoiler Location**
**Figure 1**

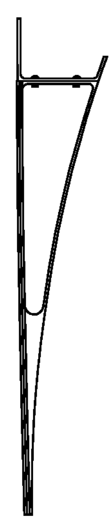
**Table 1**:

| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A4001 | Spoiler Installation |
| 113A4600 | Spoiler Assembly - 6 and 7, Inboard Ground |
| 113A4610 | Panel Bond Assembly - Spoiler 6 and 7 |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE



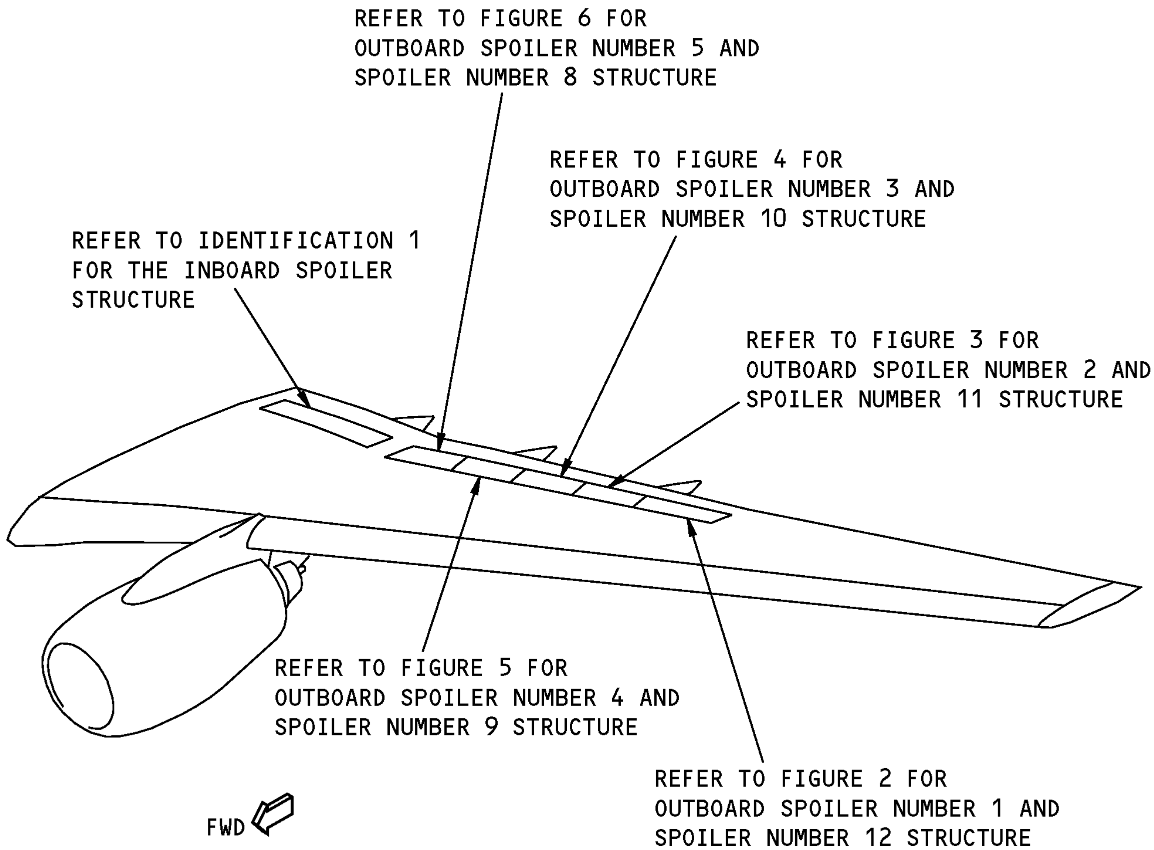
A-A

**Inboard Spoiler No. 6 and 7 Structure Identification**
**Figure 2**

**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | |
|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Leading Edge Channel | BAC1510-1354 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [2] | End Rib (2) | 7050-T7451 plate as given in AMS 4050 | |

## IDENTIFICATION 2 - OUTBOARD SPOILER STRUCTURE




NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

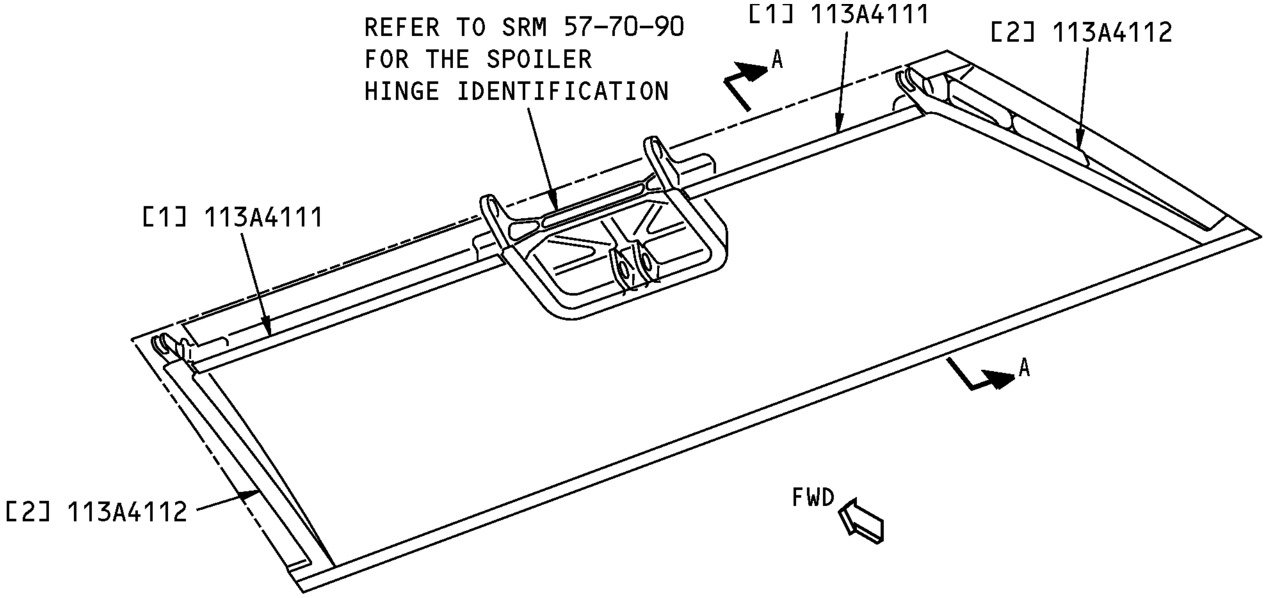
LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Outboard Spoiler Location**
**Figure 1**

**Table 1**:

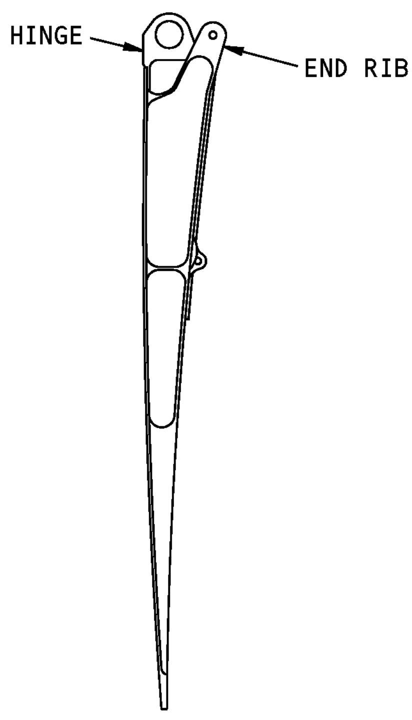
| REFERENCE DRAWINGS | |
|---|---|
| DRAWING NUMBER | TITLE |
| 113A4001 | Spoiler Installation |
| 113A4100 | Spoiler Assembly - 1 and 12, Outboard Ground |
| 113A4110 | Panel Bond Assembly - Spoiler 1 and 12 |
| 113A4200 | Spoiler Assembly - 2 and 11, Flight |
| 113A4210 | Panel Bond Assembly - Spoiler 2 and 11 |
| 113A4300 | Spoiler Assembly - 3 and 10, Flight |

| REFERENCE DRAWINGS | |
|---|---|
| **DRAWING NUMBER** | **TITLE** |
| 113A4310 | Panel Bond Assembly - Spoiler 3 and 10 |
| 113A4400 | Spoiler Assembly - 4 and 9, Flight |
| 113A4410 | Panel Bond Assembly - Spoiler 4 and 9 |
| 113A4500 | Spoiler Assembly - 5 and 8, Flight |
| 113A4510 | Panel Bond Assembly - Spoiler 5 and 8 |




NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

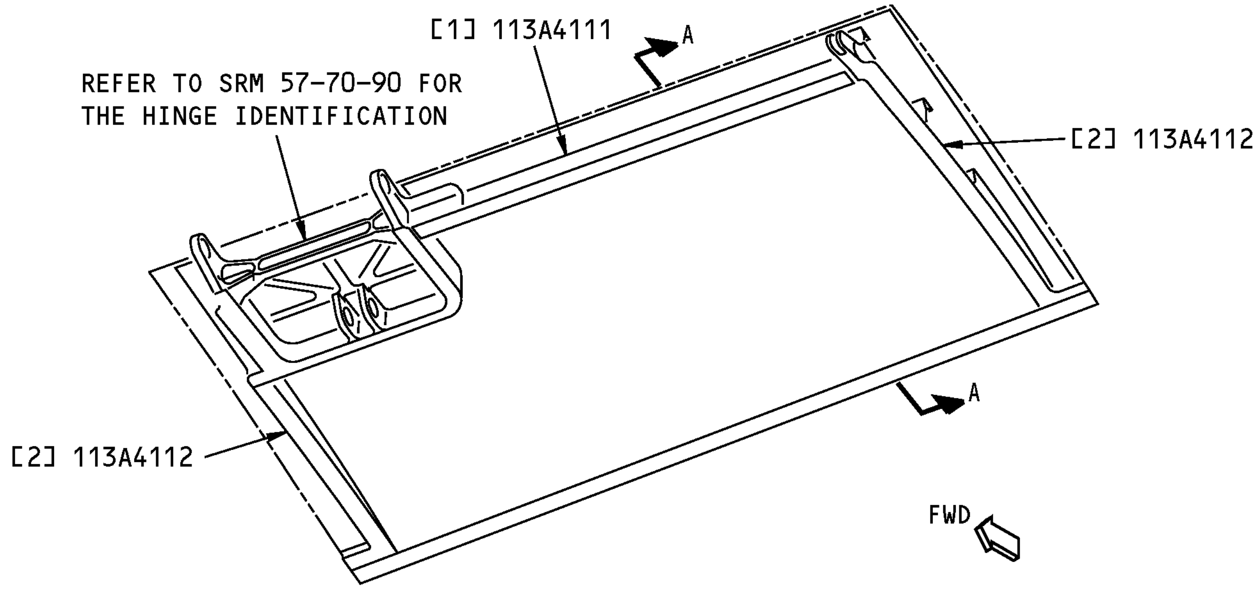



A-A

**Outboard Spoiler No. 1 and 12 Structure Identification**
**Figure 2**

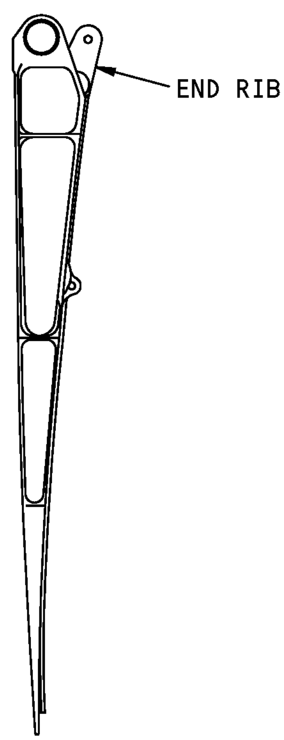
**Table 2**:

| LIST OF MATERIALS FOR FIGURE 2 | | | |
|---|---|---|---|
| ITEM | DESCRIPTION | MATERIAL | EFFECTIVITY |
| [1] | Leading Edge Channel (2) | BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [2] | End Rib (2) | 7050-T7451 plate as given in AMS 4050 | |




NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

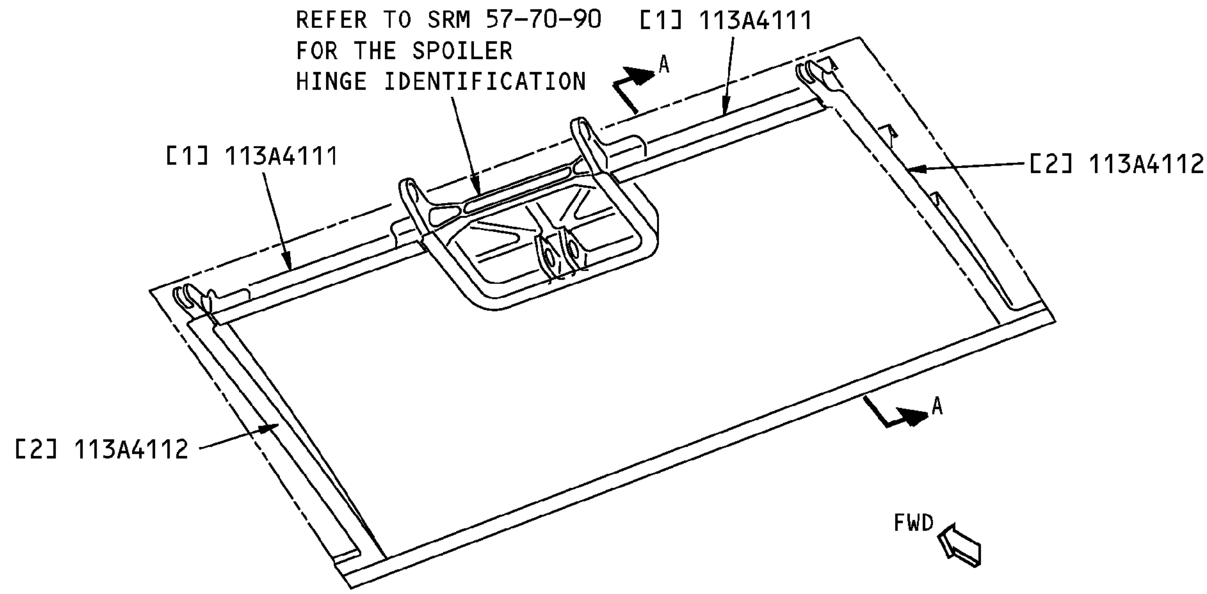



A-A

**Outboard Spoiler No. 2 and 11 Structure Identification**
**Figure 3**

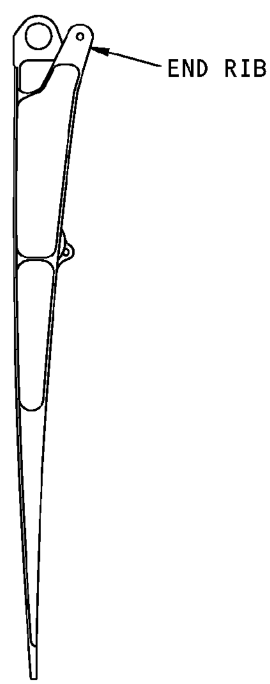
**Table 3**:

| LIST OF MATERIALS FOR FIGURE 3 | | | |
|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Leading Edge Channel (2) | BAC1510-1345 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [2] | End Rib (2) | 7050-T7451 plate as given in AMS 4050 | |




NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

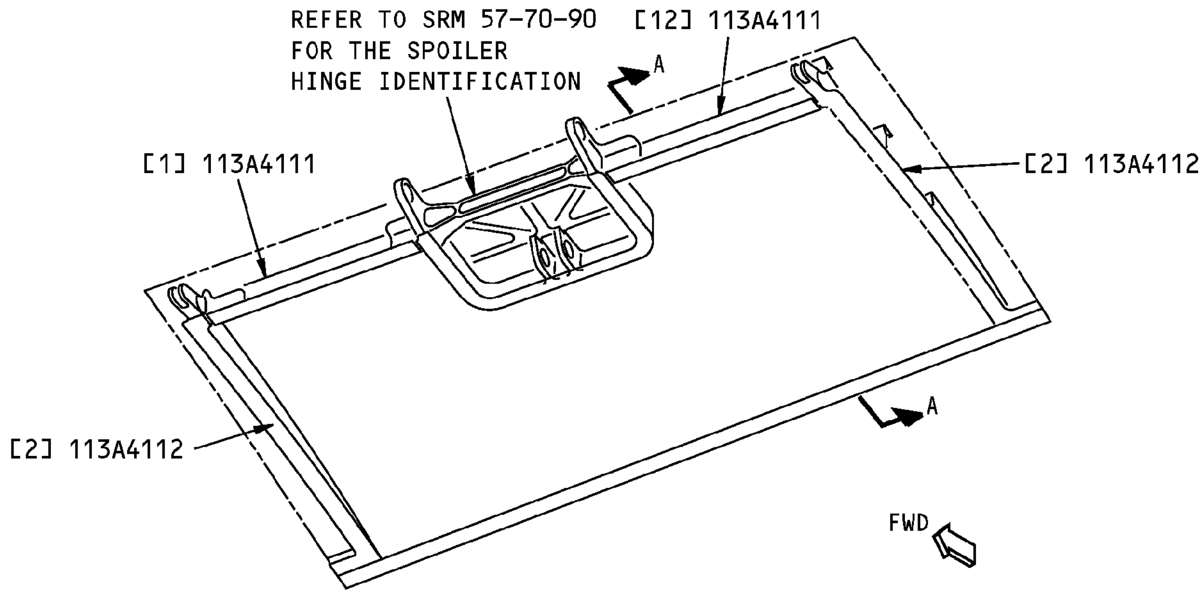



A-A

**Outboard Spoiler No. 3 and 10 Structure Identification**
**Figure 4**

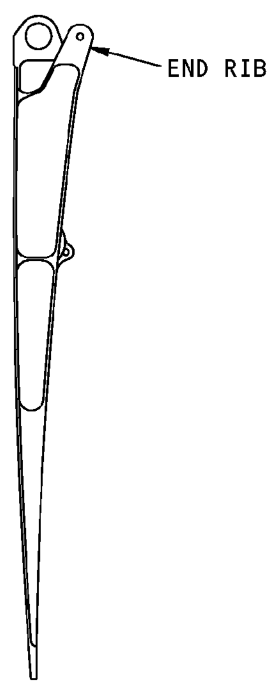
Table 4:

| LIST OF MATERIALS FOR FIGURE 4 | | | | |
|---|---|---|---|---|
| ITEM | DESCRIPTION | T | MATERIAL | EFFECTIVITY |
| [1] | Leading Edge Channel (2) | | BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [2] | End Rib (2) | | 7050-T7451 plate as given in AMS 4050 | |




NOTE: REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

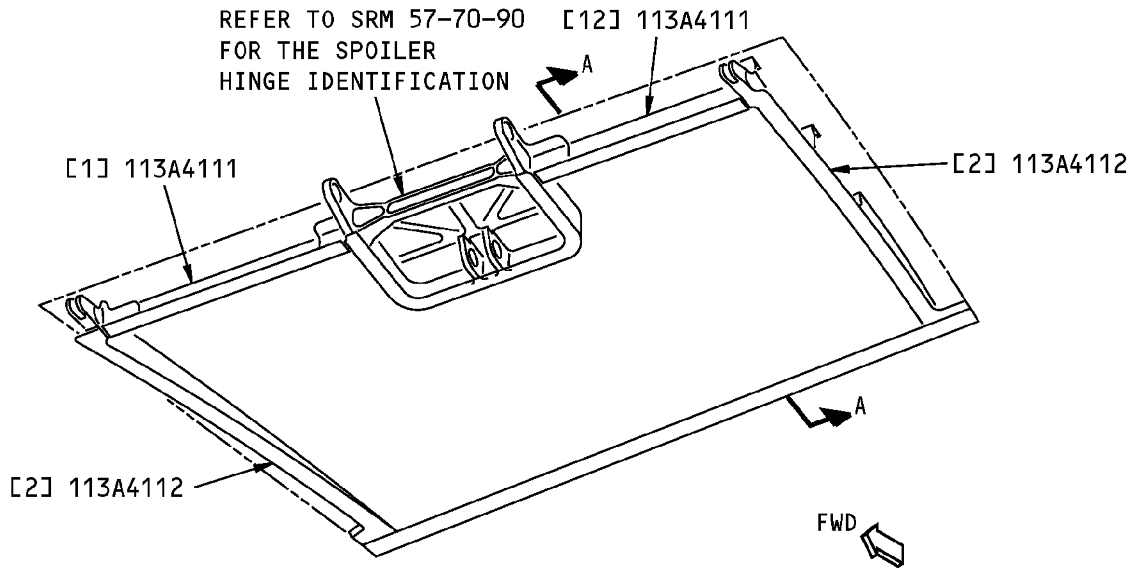



A-A

**Outboard Spoiler No. 4 and 9 Structure Identification**
**Figure 5**

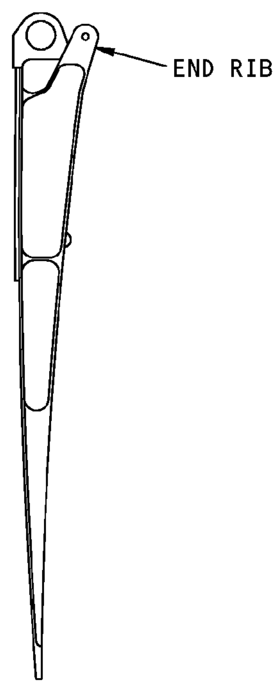
**Table 5**:

| LIST OF MATERIALS FOR FIGURE 5 | | | |
|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Leading Edge Channel (2) | BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [2] | End Rib (2) | 7050-T7451 plate as given in AMS 4050 | |




NOTE: REFER TO TABLE 6 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE




A-A

**Outboard Spoiler No. 5 and 8 Structure Identification**
**Figure 6**

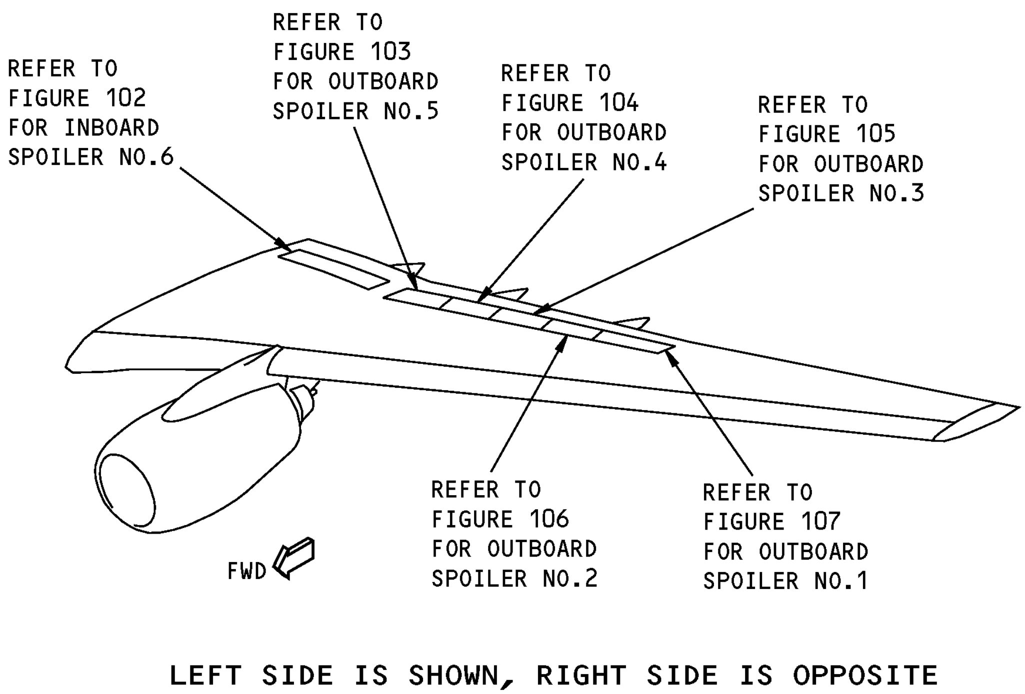
**Table 6**:

| LIST OF MATERIALS FOR FIGURE 6 | | | |
|---|---|---|---|
| **ITEM** | **DESCRIPTION** | **MATERIAL** | **EFFECTIVITY** |
| [1] | Leading Edge Channel (2) | BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11 | |
| [2] | End Rib (2) | 7050-T7451 plate as given in AMS 4050 | |

## ALLOWABLE DAMAGE 1 - INBOARD AND OUTBOARD SPOILER STRUCTURE

1. **Applicability**

   A. This subject gives the allowable damage limits for the inboard and outboard spoiler structure shown in Inboard and Outboard Spoiler Structure Locations, Figure 101/ALLOWABLE DAMAGE 1.




**Inboard and Outboard Spoiler Structure Locations**
**Figure 101**

2. **General**

   A. For each spoiler, refer to the figures that follow.

      (1) Inboard Spoiler No.6 Structure, Figure 102/ALLOWABLE DAMAGE 1, for Inboard Spoiler Number 6

      (2) Outboard Spoiler No. 5 Structure, Figure 103/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 5

(3) Outboard Spoiler No.4 Structure, Figure 104/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 4

(4) Outboard Spoiler No.3 Structure, Figure 105/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 3

(5) Outboard Spoiler No.2 Structure, Figure 106/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 2

(6) Outboard Spoiler No.1 Structure, Figure 107/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 1.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

C. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

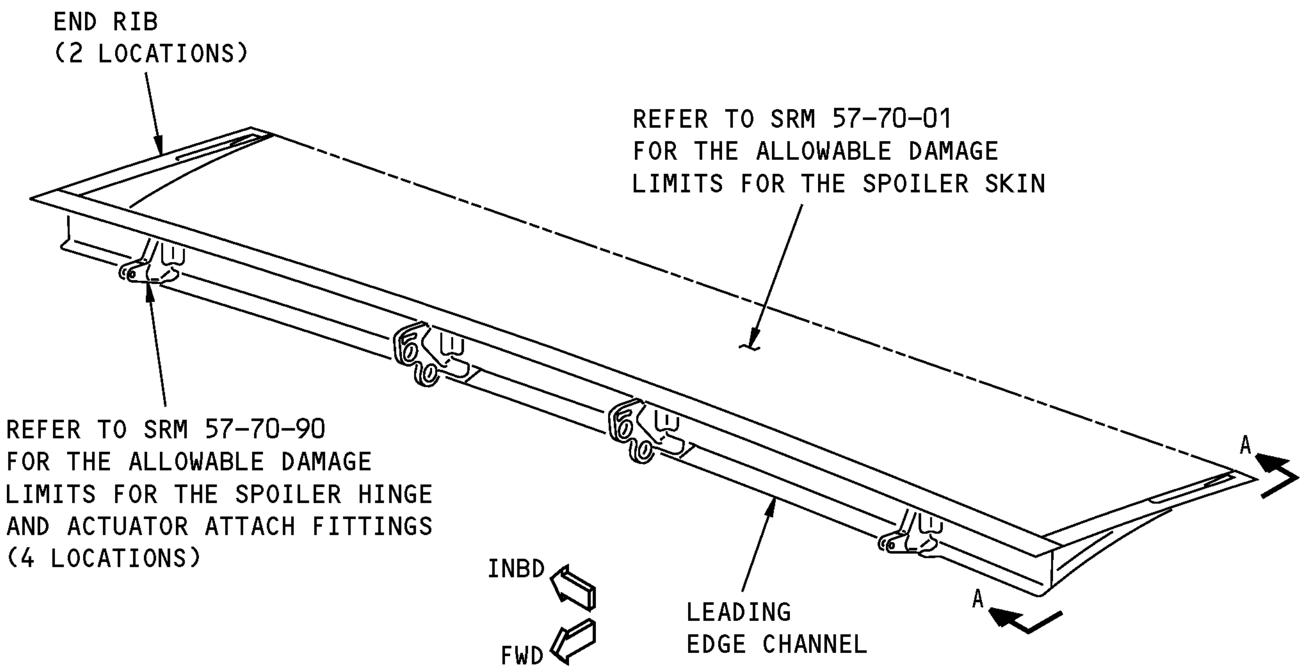
(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

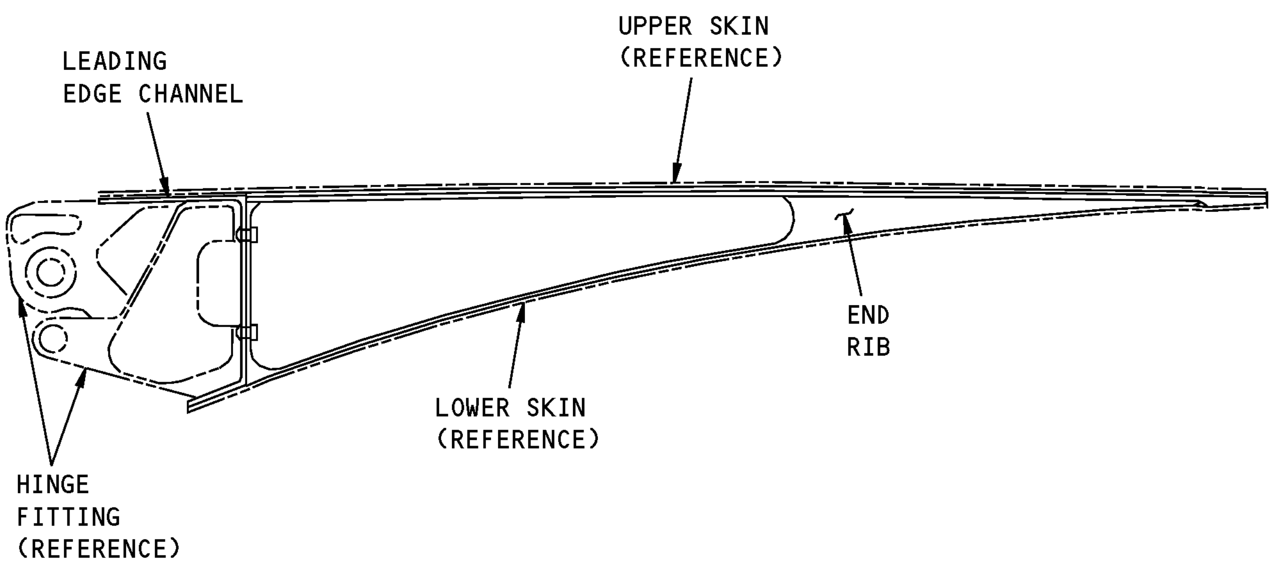
D. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(2) Apply one layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-41-02.

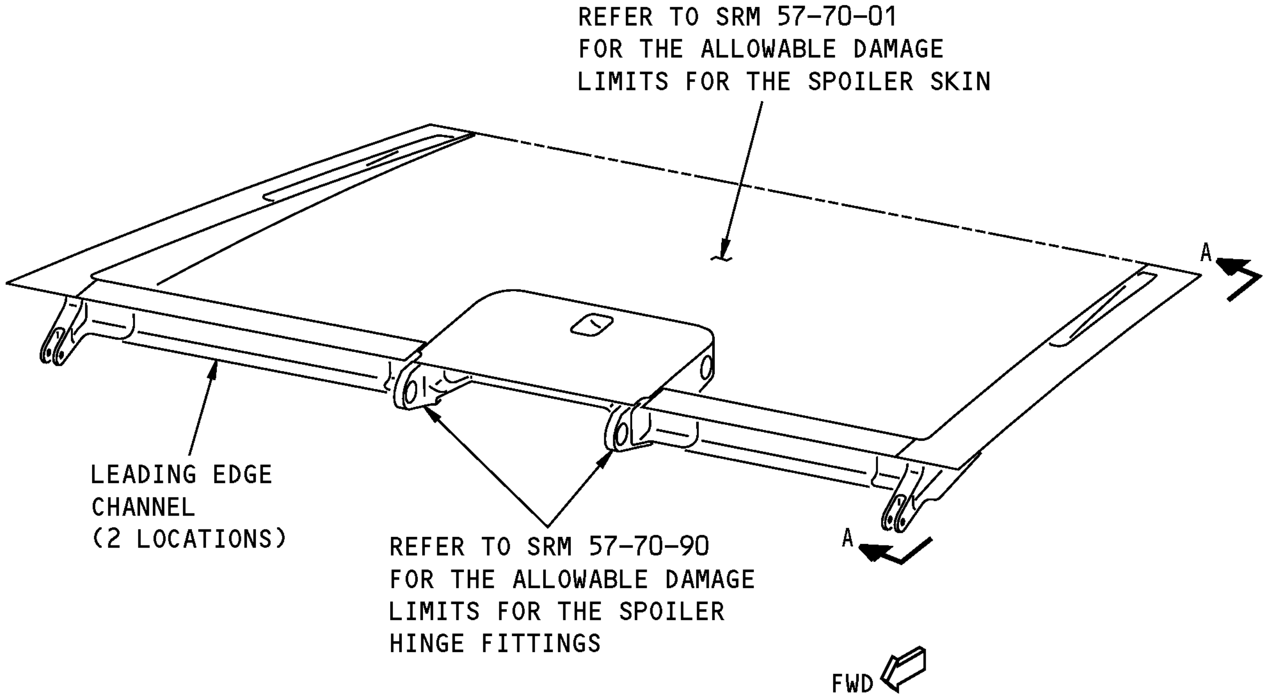



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

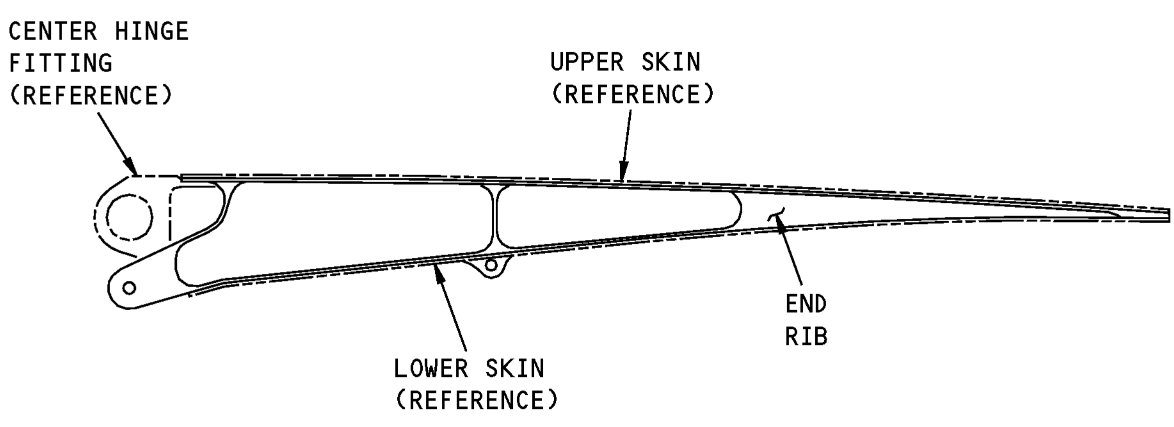



A-A

**Inboard Spoiler No.6 Structure**
**Figure 102**

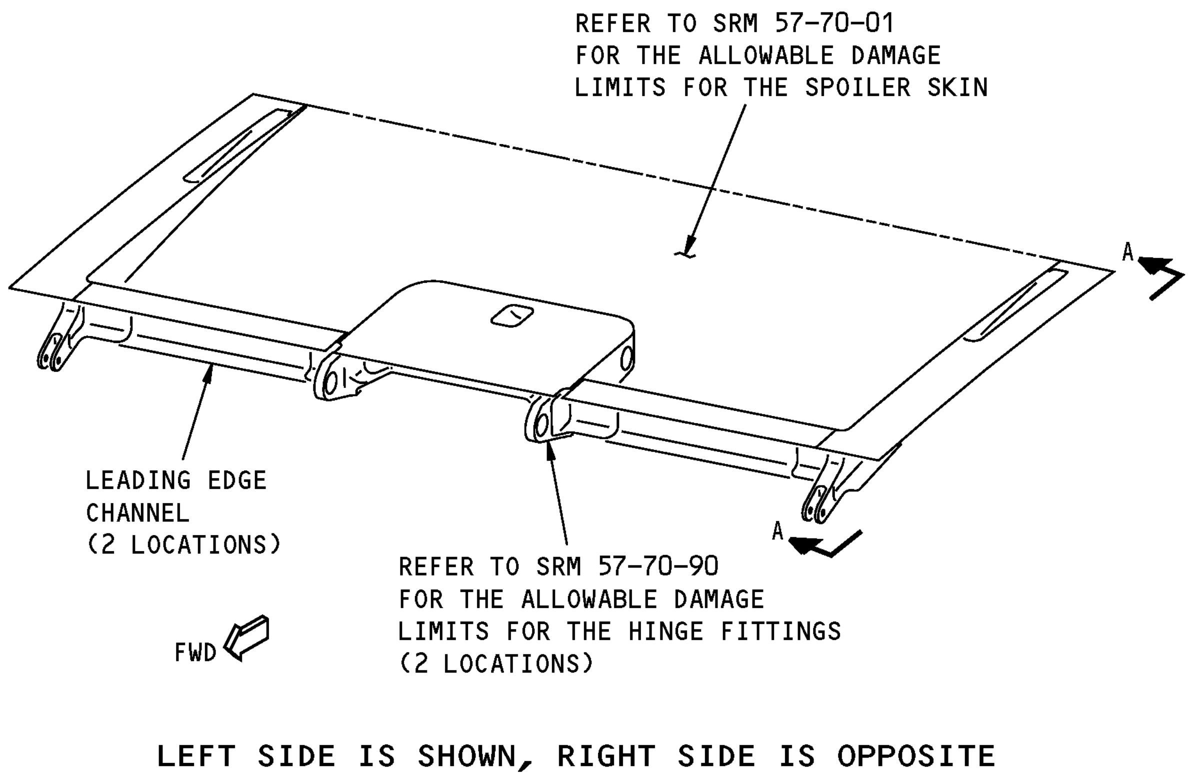


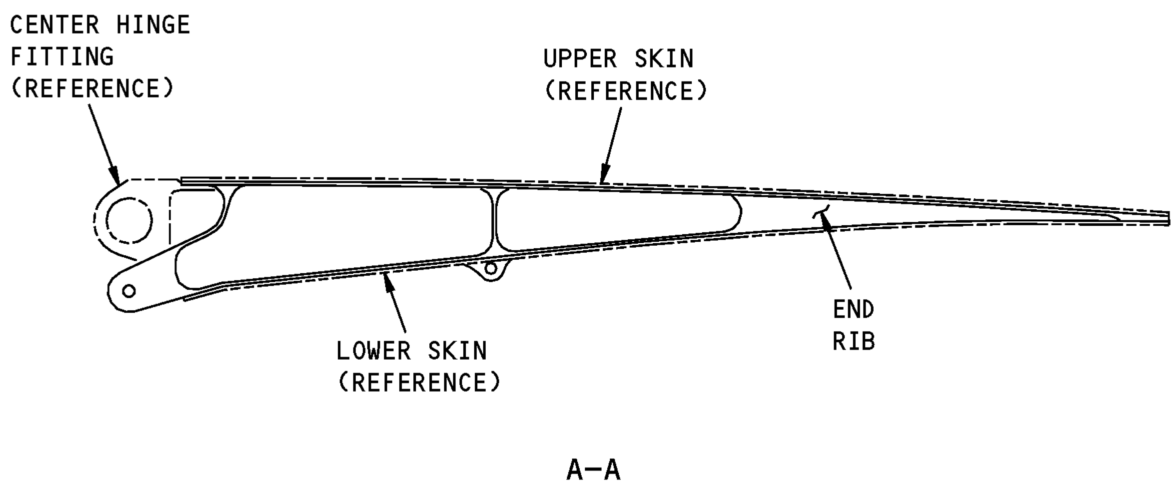

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE



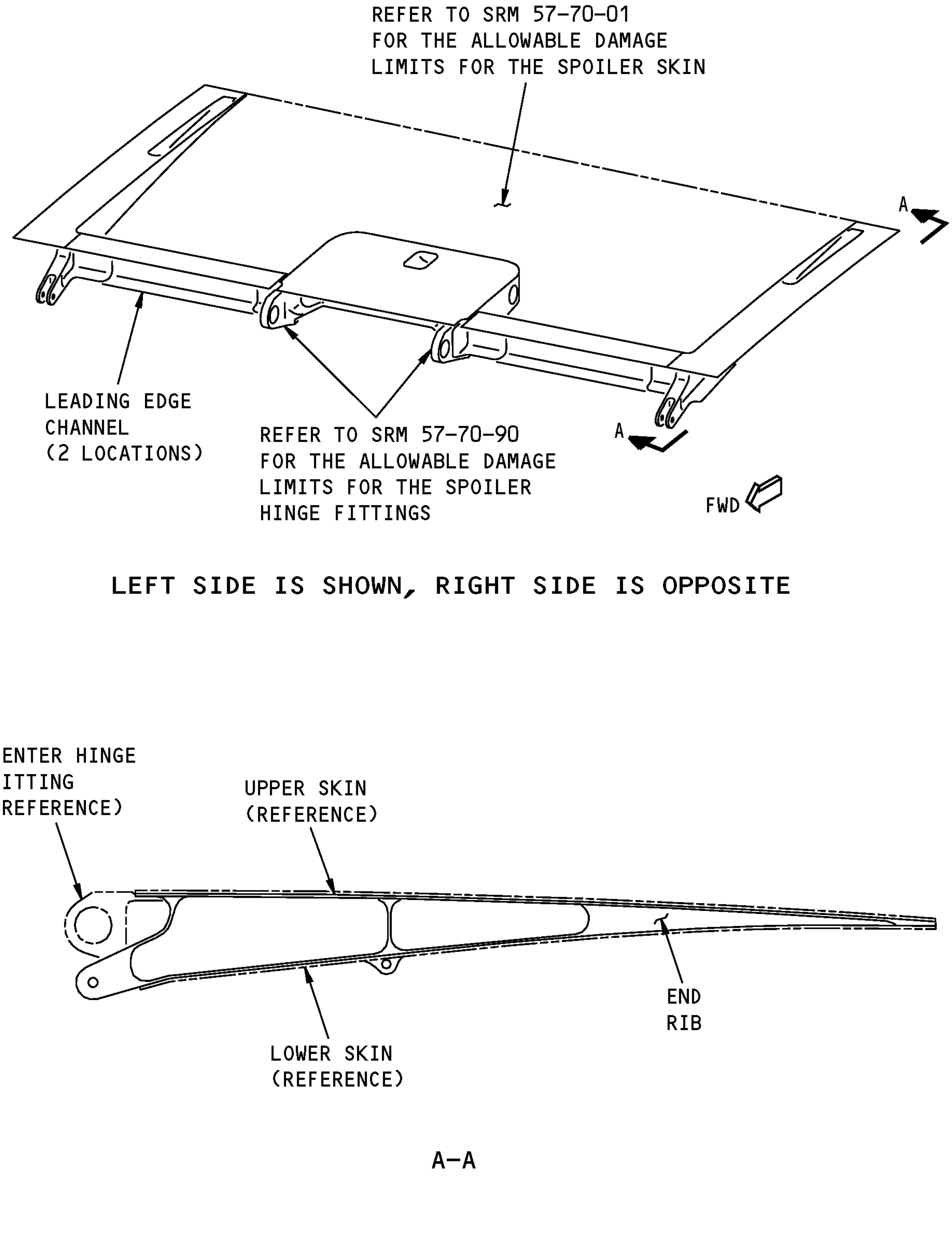

A-A

**Outboard Spoiler No. 5 Structure**
**Figure 103**

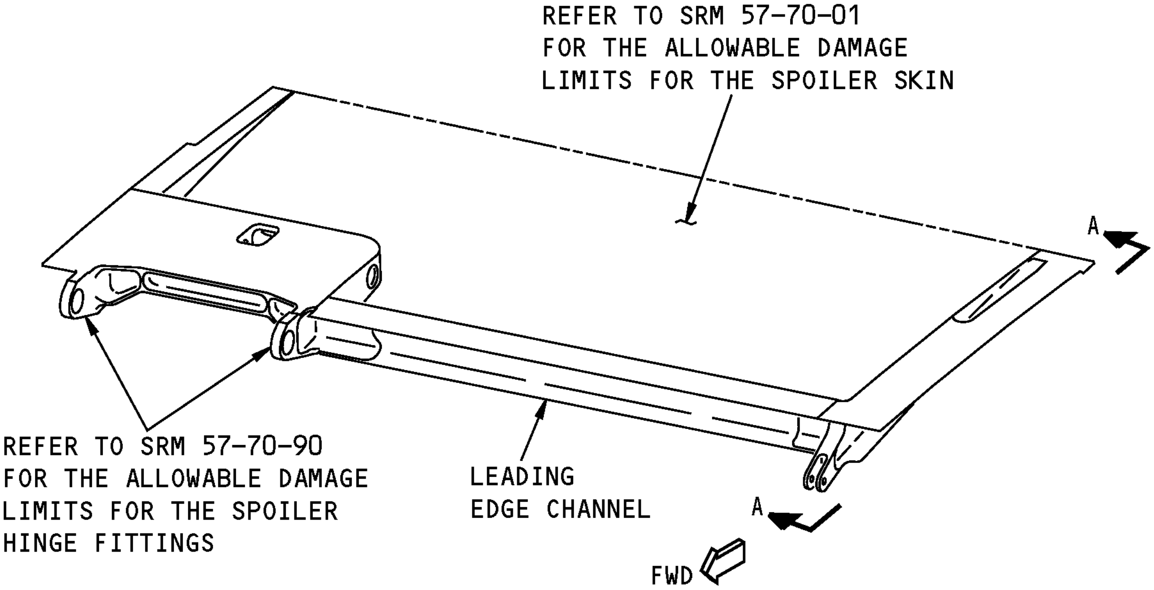


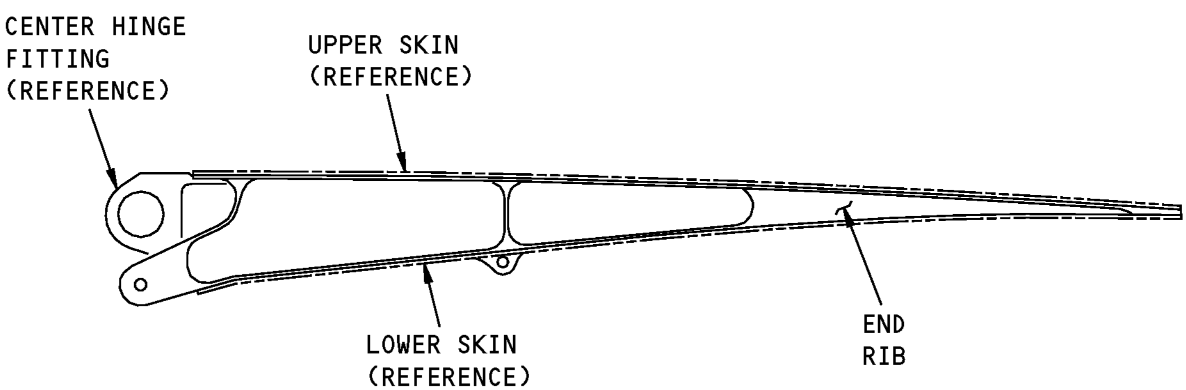

**Outboard Spoiler No.4 Structure**
**Figure 104**




**Outboard Spoiler No.3 Structure**
**Figure 105**

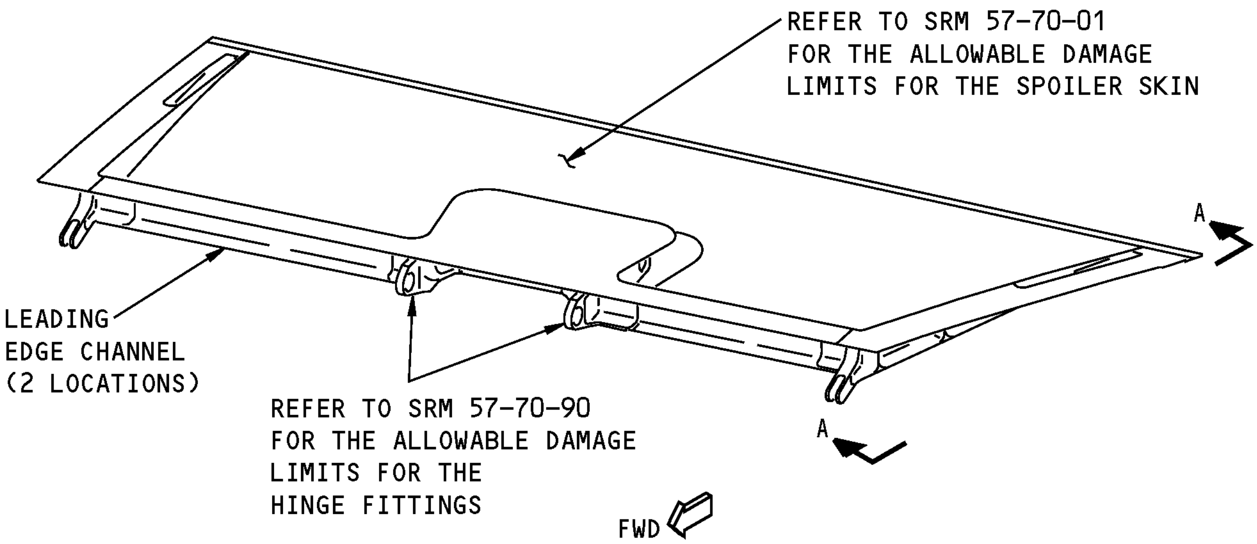



LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

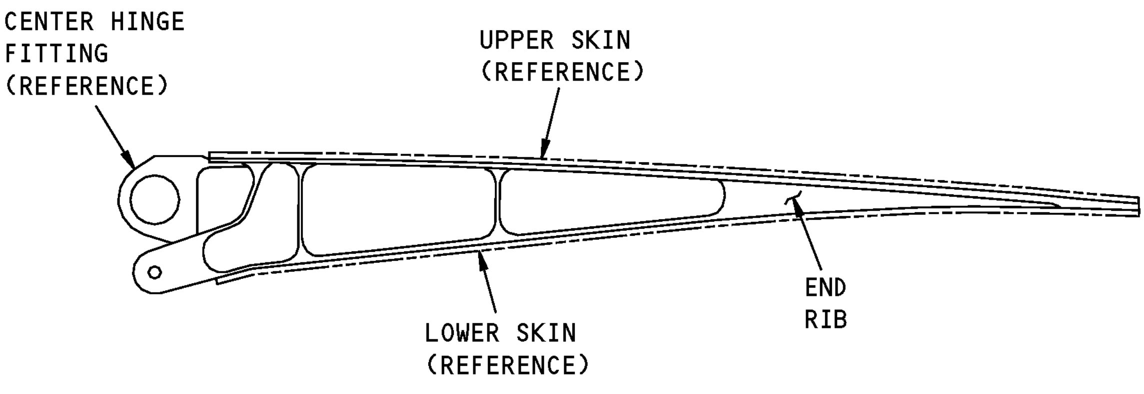



A-A

**Outboard Spoiler No.2 Structure**
**Figure 106**




LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE




A-A

**Outboard Spoiler No.1 Structure**
**Figure 107**

## 3. References

| Reference | Title |
|---|---|
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-30-03 | NON-METALLIC MATERIALS |
| 51-30-05 | EQUIPMENT AND TOOLS FOR REPAIRS |
| SOPM 20-41-02 | Application of Chemical and Solvent Resistant Finishes |

## 4. Allowable Damage Limits

A. Cracks:

(1) Remove edge damage as shown in Spoiler Structure Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , and C .

B. Nicks, Gouges, Scratches, and Corrosion:

(1) You are permitted to do one of the procedures that follows:

(a) Remove damage as shown in Spoiler Structure Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(b) Drill out the damage as given for Holes and Punctures.

**NOTE**: The damage cannot be drilled out as a hole if the damage has been blended out.

C. Dents:

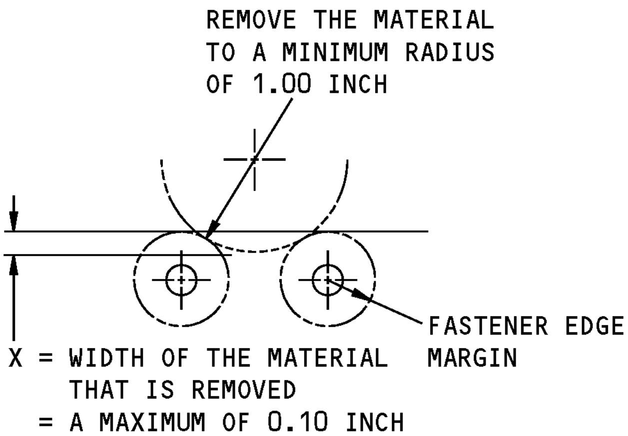
(1) The damage is permitted as shown in Spoiler Structure Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail F .

D. Holes and Punctures are permitted if they are:

(1) A maximum of 0.25 inch in diameter

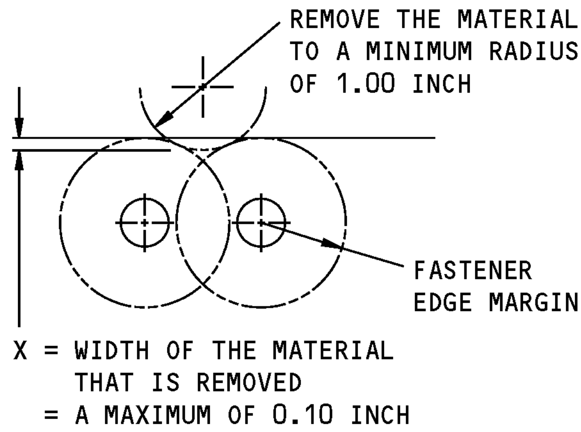
(2) A minimum of 4D (D = diameter of the damage) from other damage, fastener holes, or material edges

(3) Filled with an aluminum 2117-T3 or 2117-T4 protruding head rivet.



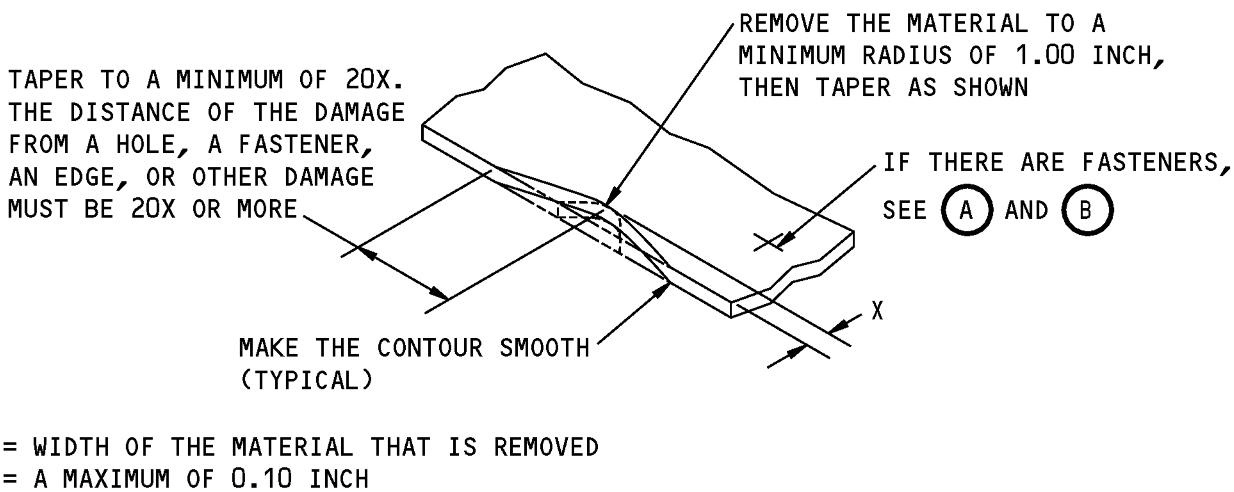

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A




REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

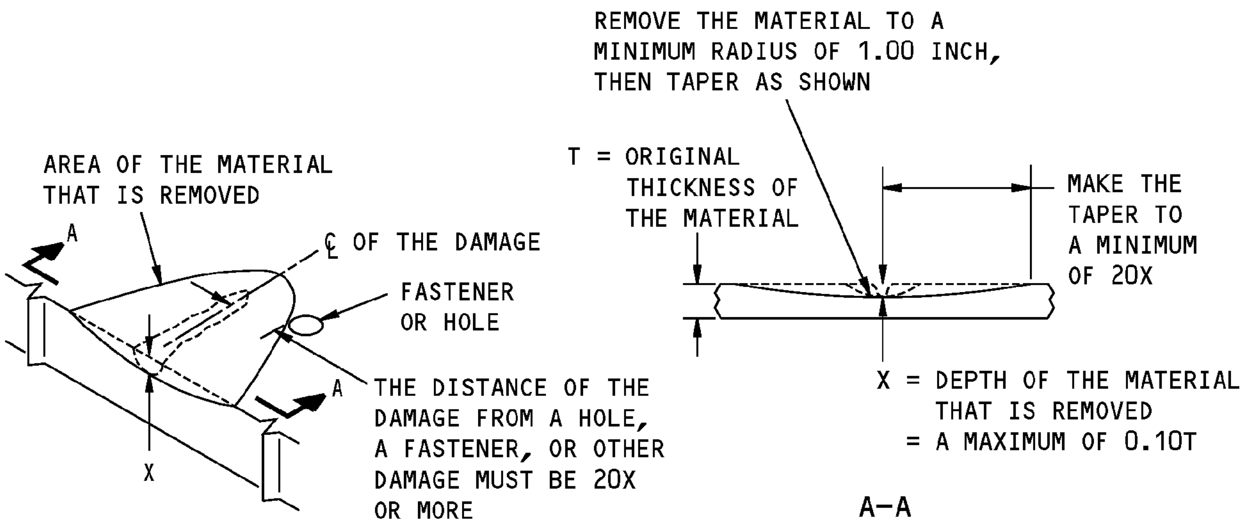
B




REMOVAL OF DAMAGED MATERIAL AT AN EDGE

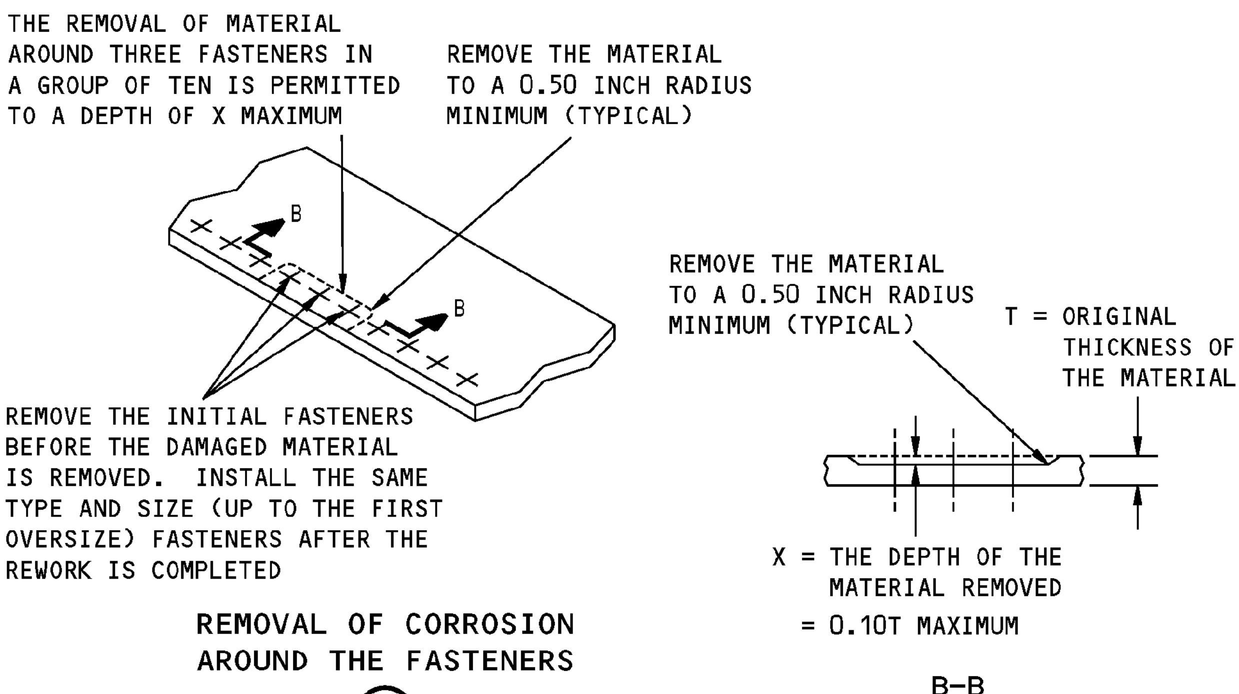
C

**Spoiler Structure Allowable Damage Limits**
**Figure 108 (Sheet 1 of 3)**




REMOVAL OF DAMAGED MATERIAL ON A SURFACE

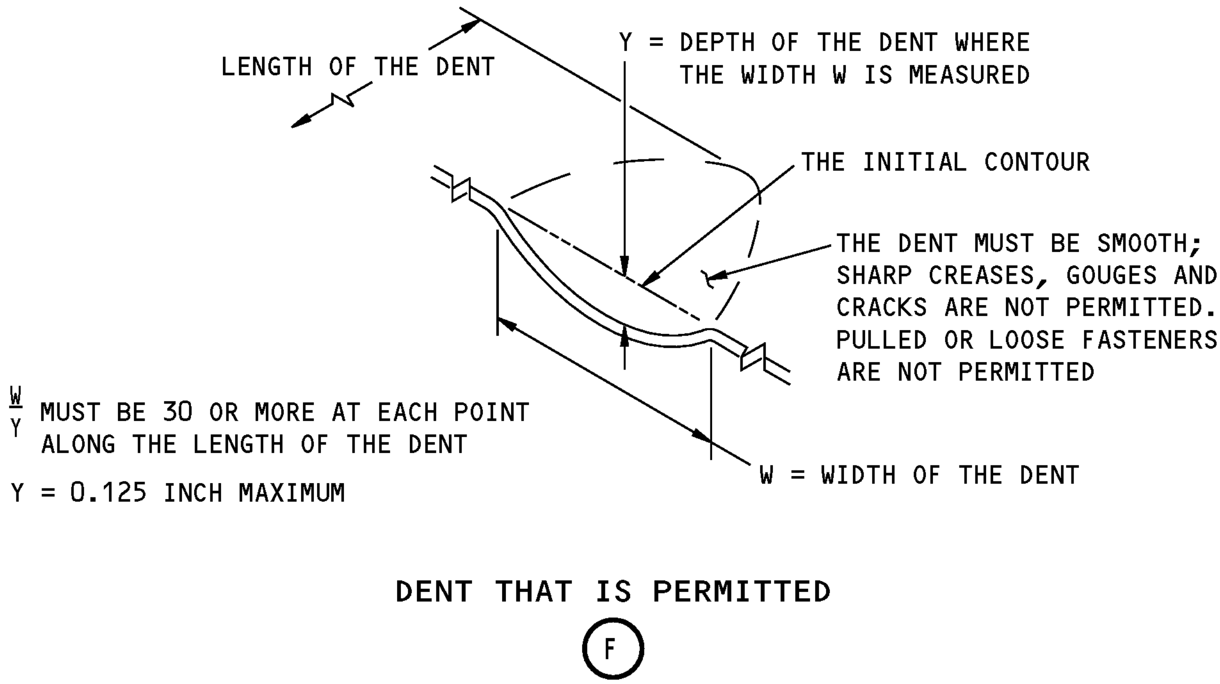
D




REMOVAL OF CORROSION AROUND THE FASTENERS

E

**Spoiler Structure Allowable Damage Limits**
**Figure 108 (Sheet 2 of 3)**

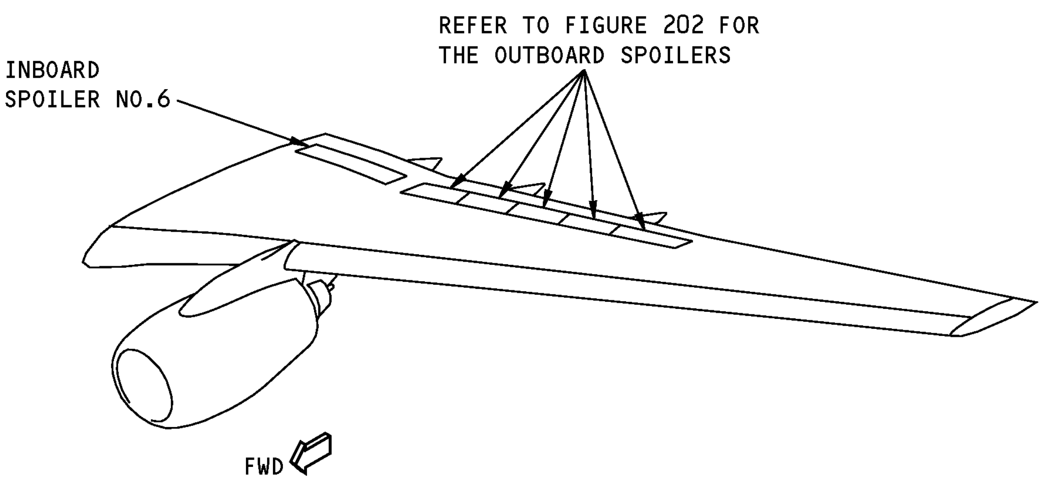



**Spoiler Structure Allowable Damage Limits**
**Figure 108 (Sheet 3 of 3)**

## REPAIR 1 - SPOILER END RIB

### 1. Applicability

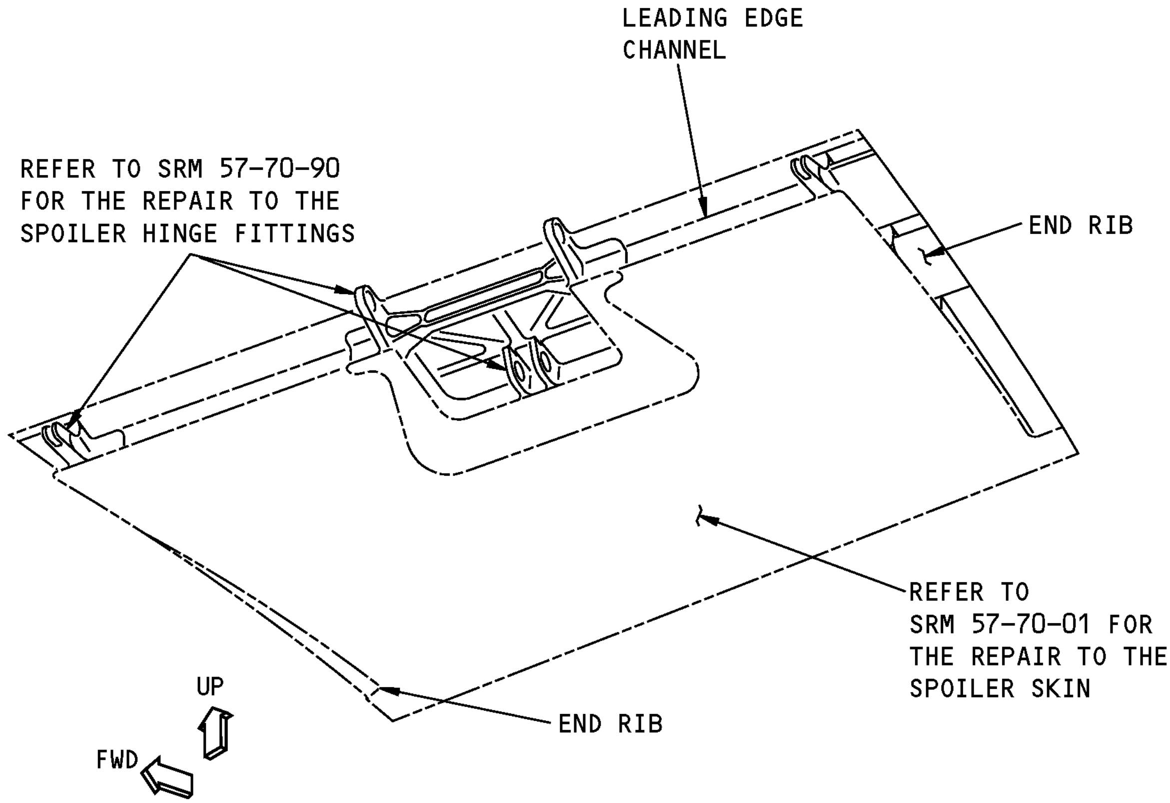
A. Repair 1 is applicable to damage to the spoiler end rib shown in Inboard and Outboard Spoiler Locations, Figure 201/REPAIR 1.




NOTE: BOEING HAS NOT FOUND IT NECESSARY TO SUPPLY REPAIRS FOR THE INBOARD SPOILER NO. 6 (LEFT AND RIGHT) IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

**Inboard and Outboard Spoiler Locations**
**Figure 201**




THE LEFT SIDE SPOILER IS SHOWN,
THE RIGHT SIDE SPOILER IS OPPOSITE
(A TYPICAL OUTBOARD SPOILER IS SHOWN)

**Typical Outboard Spoiler**
**Figure 202**

**2. General**

A. For the shapes of the spoilers, refer to the identification section.

B. Repair 1 is a Category C repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

C. Do an inspection of the blind rivet repair at each 5000 flight cycle interval.

D. Blind rivets that are loose, damaged, or that have fallen out must be replaced.

E. Repair 1 can be used only if you install the blind rivets flush against the internal structure.

**3. References**

| Reference | Title |
|---|---|
| 51-00-06 | STRUCTURAL REPAIR DEFINITIONS |
| 51-10-02 | INSPECTION AND REMOVAL OF DAMAGE |
| 51-20-01 | PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS |
| 51-20-05 | REPAIR SEALING |
| 51-40-02 | FASTENER INSTALLATION AND REMOVAL |
| 51-40-03 | FASTENER SUBSTITUTION |
| 51-40-05 | FASTENER HOLE SIZES |
| 51-40-06 | FASTENER EDGE MARGINS |
| AMM 27-61-11 | FLIGHT SPOILER |
| AMM 27-62-91 | OUTBOARD GROUND SPOILER |
| AMM 51-21-99 P/B 701 | DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING |
| AMM 27-61-11 | Flight Spoiler |
| AMM 27-62-12 | Inboard Ground Spoiler Removal and Installation |
| SOPM 20-20-02 | Penetrant Methods of Inspection |
| SOPM 20-44-04 | Application of Urethane Compatible Primers |
| 737 NDT Part 6, 51-00-00 | Structures - General |
| 737 NDT Part 6, 51-00-00, Figure 4 | Surface Inspection of Aluminum Parts |

**4. Repair Instructions**

A. Get access to the damaged area.

(1) If necessary, remove the spoiler from the wing. Refer to the sections that follow:

(a) AMM 27-62-12 for the Inboard Spoiler Number 6

(b) Refer to AMM 27-62-91/401 for the Outboard Spoiler Number 1

(c) AMM 27-61-11 for the Outboard Spoiler Numbers 2, 3, 4, and 5.

B. Remove the damage to a maximum of 50 percent of the initial part thickness. Refer to 51-10-02.

(1) If necessary, cut out the damaged area.

C. Do a 10X visual inspection of the area to make sure that all damaged material is removed.

(1) It is optional to do an eddy current inspection. Refer to 737 NDT Part 6, 51-00-00, Figure 4 .

(2) If corroded material was removed, it is optional to do a dye penetrant inspection. Refer to SOPM 20-20-02.

D. Make the repair parts as shown in Spoiler End Rib Repair, Figure 203/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair materials.

**Table 201**:

| REPAIR MATERIAL | | | |
|---|---|---|---|
| ITEM | PART | QUANTITY | MATERIAL |
| [1] | Angle | 1 | Use 2024-T4 extrusion that is one gage more than the rib flange |
| [2] | Strap | 1 | Use bare or clad 2024-T3 that is one gage more than the rib flange |
| [3] | Filler | As necessary | Use bare or clad 2024-T3 that is the same gage as the material that was removed |

E. Assemble the repair parts.

F. Drill the fastener holes.

G. Remove the repair parts.

H. Remove all nicks, scratches, burrs, and sharp edges from the repair parts.

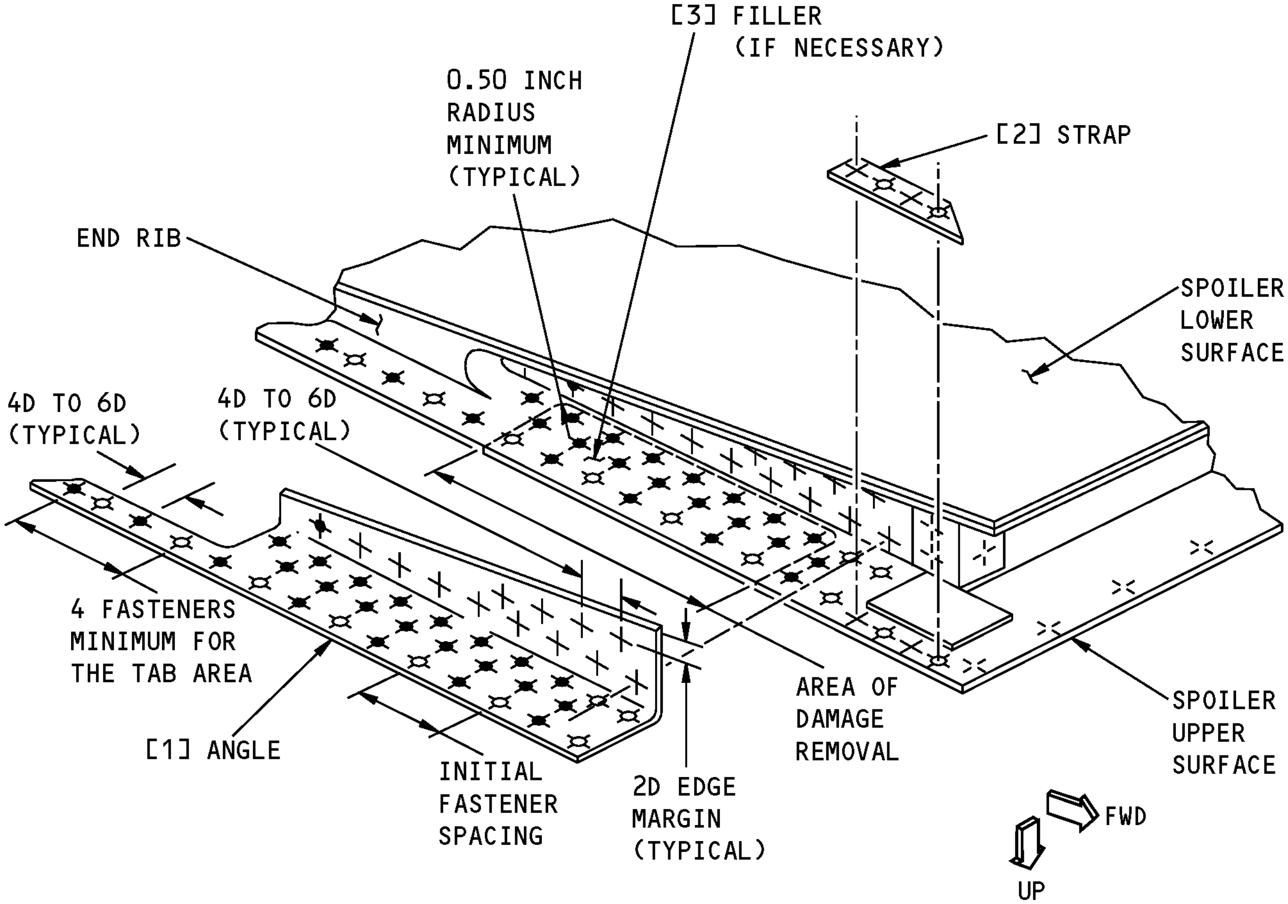
I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the rib flange.

J. Apply one layer of BMS 10-79, Type II primer to the repair parts and to the bare edges of the rib flange. Refer to 51-20-01.

K. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

L. Install the fasteners without sealant.

M. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.




NOTES

- D = THE DIAMETER OF THE FASTENER.

FASTENER SYMBOLS

-¦- REFERENCE FASTENER LOCATION.

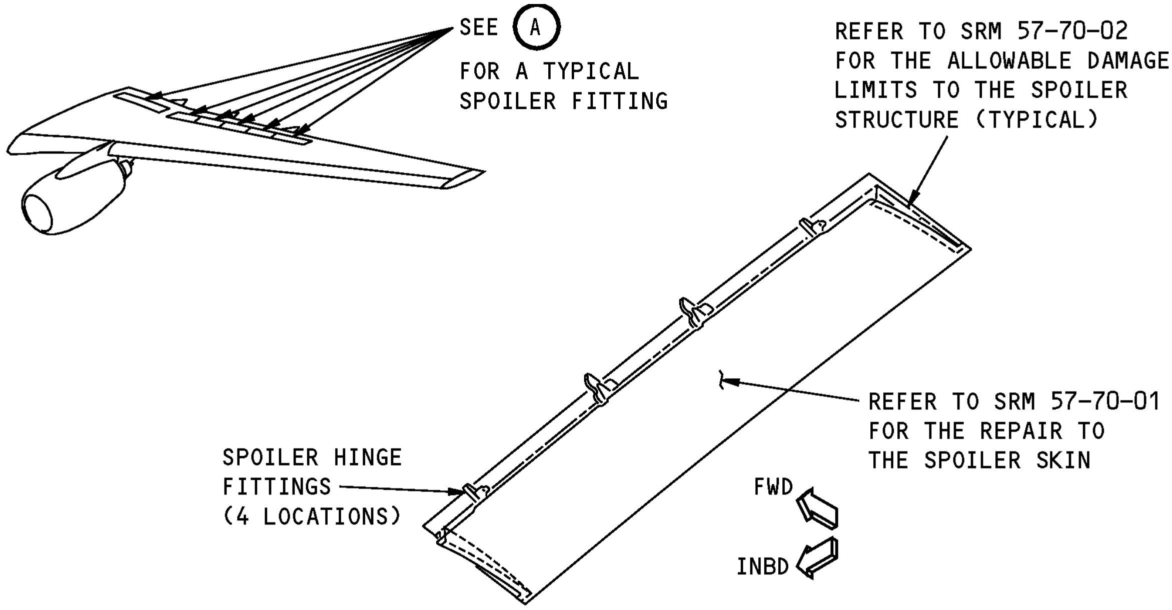
-◇- INITIAL FASTENER LOCATION. INSTALL A BACR15FT5 RIVET OR AN MS20470DD5 RIVET.

-◆- REPAIR FASTENER LOCATION. INSTALL A BACR15FT5 RIVET OR AN MS20470DD5 RIVET.

+ REPAIR FASTENER LOCATION. INSTALL AN NAS1398B BLIND RIVET.

**Spoiler End Rib Repair**
**Figure 203**

## ALLOWABLE DAMAGE 1 - WING SPOILER FITTINGS




NOTE: FOR THE ALLOWABLE DAMAGE LIMITS TO THE SPOILER FITTINGS, REFER TO:

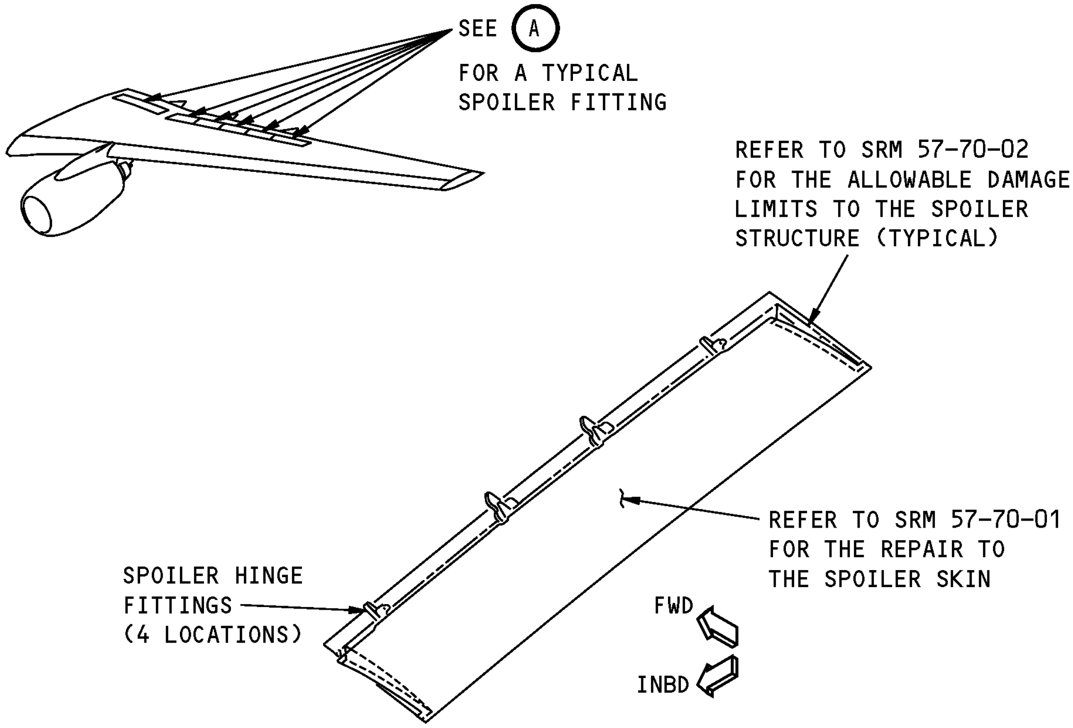
- CMM 57-56-63 FOR OUTBOARD SPOILER NUMBER 1
- CMM 57-56-53 FOR OUTBOARD SPOILER NUMBER 2
- CMM 57-56-54 FOR OUTBOARD SPOILER NUMBER 3
- CMM 57-56-55 FOR OUTBOARD SPOILER NUMBER 4
- CMM 57-56-56 FOR OUTBOARD SPOILER NUMBER 5
- CMM 57-56-64 FOR INBOARD SPOILER NUMBER 6

LEFT SIDE SPOILER IS SHOWN, RIGHT SIDE SPOILER IS OPPOSITE

A

**Typical Spoiler Fitting**
**Figure 101**

## REPAIR 1 - WING SPOILER FITTINGS




NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE SPOILER IS SHOWN, RIGHT SIDE SPOILER IS OPPOSITE

A

**Typical Spoiler Fitting Repair**
**Figure 201**