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Repair of Advanced Composite Structures

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This paper presents the concluding results of the large area composite structure repair program in which five large panels were repaired and subjected to static tests. A description of the Northrop bonded-cocured scarf joint repair is given with a review of strength and durability test results. Application of this technique to the problem presented in each of five panels is discussed. Finally, the repair of a speedbrake damaged in service is described. Repairability of advanced composite structure is fully demonstrated.

Introduction

THE common problem of damaged aircraft structure confronts the commercial user with considerable economic loss if no adequate repair procedure is known. On the other hand, with the damage done (an inevitability) we have opportunity for considerable gain by salvaging the structure instead of replacing it, if strong, durable repairs can be made at a reasonable cost. The military aircraft operator may be faced with a problem more serious than the economic loss. "For the want of a nail the shoe is lost . . .."

The objective of any repair is to restore the part to its original strength and durability, though a lower strength can often be justified. Current (i.e., 1979) advanced composite structural criteria establish ultimate load strain levels at approximately 4000 μin./in. to allow for strength degradation due to fastener holes and to prevent matrix cracking. Hence, a repair which can develop 4000 μin./in. might be considered adequate for strength (i.e., ultimate load is supported), however, it is unlikely that the original strength and durability of the part would be restored.

The objective of the large area composite structure repair program[1] has been to develop repair techniques that would restore as much as possible of the original material strength. Hence, the room temperature tension target value strain was 11,000 μin./in., or 2.75 times the typical ultimate load strain of 4000 μin./in. The ultimate allowables for AS/3501-5 graphite/epoxy, shown in Table 1 are the reference values for calculating joint efficiencies of the 16-ply $[(\pm 45/0/90)_2]_s$ laminate repairs in the joint development tests of Ref. 1.

Patch material was AS/3501-6 graphite/epoxy prepreg (35% resin) cured under vacuum pressure at 350°F for two hours with a no-bleed layup procedure. The FM-400 (or RB 398) film adhesive was cocured with the patch onto the parent laminate. The parent had been conditioned to 1% average moisture content to represent the service condition of real parts. These repair materials were chosen to meet a maximum service temperature requirement of 265°F.

Northrop Bonded-Cocured Scarf Repair

The basic concept of the scarf joint is obvious and simple. As the thickness of one side of the joint (the parent laminate) is diminished, the thickness of the opposite side (the patch) is increased. Load is transferred through adhesive shear only since the scarf length is large compared to the laminate thickness.

An elementary procedure was used for load transfer analysis based on the assumption that at any station the load in each adherend is proportional to the extensional stiffness. This procedure accounts for the heterogeneous nature of the laminate and gives a satisfactory description of the shear distribution in the splice when applied at each ply end in the splice. The analytical procedure and a discussion of the development of the Northrop bonded-cocured scarf joint may be found in Ref. 1 or 2.

A real repair joint is not like the idealization of a scarf joint in a number of respects. Most importantly, the patch is cocured, and hence the scarf on the patch is approximated by steps as each ply is succeedingly longer proceeding outward. Cocuring eliminates fit problems since the uncured plies mold readily to the shape of the tooling surface defining the inside mold line, the scarf and the outer surface.

The Northrop bonded-cocured scarf repair is shown in Fig. 1. The shortest patch ply overlaps the end of the parent laminate by 0.20 in. with each subsequent ply end 0.10 in. or more longer. The two longest 0-deg ply ends are extended to a 0.30-in. step and serrated with standard pinking shears a depth of 1/8 in. In addition to replacement plies arranged in the same stacking order as the parent laminate, one ply in each of three directions (0, 45, −45 deg) has been added to each side. The added plies make up for the lower strength in the vacuum pressure cured patch laminate, about 84% of the parent. The added ±45 deg plies are essential for shear strength restoration. The inner surface plies keep the patch aligned with the load and assure adequate load transfer from the end of the parent laminate, which in reality may not be well scarfed. The short ply on the inner patch appears in retrospect to be unnecessary.

The same joint design approach was applied to a laminate with an alternate stacking order $[(\pm 45/0/90)_2]_s$ with equally successful results. The patch stacking order of the replacement plies was made identical to the parent laminate. The test program discussed in detail in Ref. 1 was conducted largely with this alternate stacking order.

Strength and Durability Data

The strength and durability of the Northrop bonded-cocured scarf repair was thoroughly examined in a series of 16 test conditions, the major categories of which are 1) dry or wet and 2) tension or compression. Each quadrant of the series (e.g., dry, tension) consists of static tests at −65°F, room temperature, and 265°F plus two fatigue loaded groups subsequently subjected to static test at room temperature. The fatigue loading was a randomized spectrum for the F-5 wing root lower surface, with compression loads truncated, imposed for two airplane lifetimes or 8000 flight hours. The

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Table 1 Parent laminate allowables[a]

	Tension allowable		Compression allowable		Shear allowable
Temp, °F	Loading, lb/in.	Strain, μin./in.	Loading, lb/in.	Strain, μin./in.	Loading, lb/in.
−65	6210	–	9740	–	–
RT	6860	11,000	7600	12,000	2836
265	5750	–	4710	–	–

[a]Cured at 100 psi/350° F/2 hour; laminate stacking $[(\pm 45/0/90)_2]_s$; moisture content = 1.0%. B-basis tension and shear allowables from Ref. 3; compression allowables are 80% of average of 3 tests.

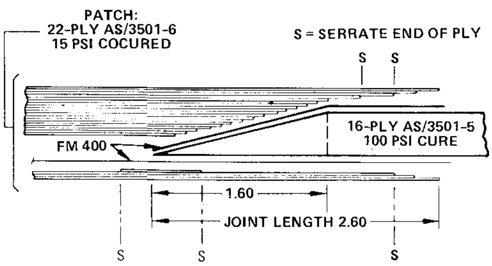

Fig. 1 Northrop bonded-cocured scarf joint repair.

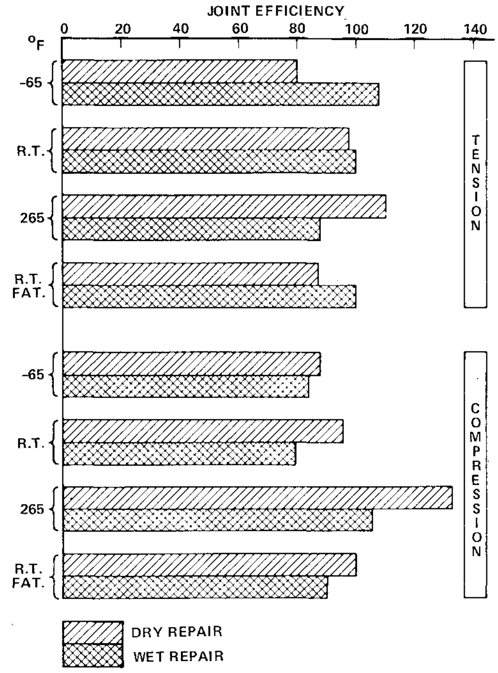

Fig. 2 Northrop bonded-cocured scarf joint results.

maximum load in the spectrum was scaled to produce a peak strain of 4570 μin./in. one group and 2510 μin./in. in the second group. Wet repairs were moisture conditioned at 140°F/95% relative humidity (RH) for 30 days.

The results are shown graphically in Fig. 2 where each bar represents the average of three tests. Those with fatigue history are represented by the lower result of the two groups. No indication of failure was seen during fatigue loading. The prior fatigue history appears to have little effect on the static strength of the repair in either tension or compression.

Moisture conditioning of the repaired assembly deteriorated the splice somewhat as shown by the compression results. Splice-related failures were more frequent among the wet specimens, both tension and compression, also indicating deterioration. Failure load results of both wet and dry repairs are compared to the same allowable from Table 1. The lowest result in Fig. 2 is 79% for the compression loaded, room temperature wet group.

The results of Fig. 2 are part of a larger program of 356 individual tests on 12 different specimen types, each in some way related to the bonded scarf joint described here. This data was largely reported in Ref. 2. The conclusion was clearly that the Northrop bonded-cocured scarf repair can reliably develop 80% of the B-basis allowable fiber failure strength even under adverse environmental conditions and appeared to be applicable to various repair situations. The final requirement was to demonstrate this applicability.

Large Area Damage Repair Demonstrations

The demonstration components were five panels, 19×60 in., representing both lightly-loaded and highly-loaded types of aircraft structure and each presenting a different repair problem. Each panel was repaired, loaded in various ways, and finally failed in tension. The panels were tested as the lower cover in the central section of the box beam of Fig. 3. Since each panel was mechanically fastened to the spars of the box beam with 3/16-in. fasteners, the parent laminate allowable could never be realized in the panel. The work of Verette and Labor[3] shows that unloaded 3/16-in. flush head fastener holes in a $[(0/\pm 45/90)_2]_s$ laminate reduce the panel ultimate tensile strength to one-half of the strength without holes and that only a laminate with a much larger percentage of ±45-deg plies will result in a higher relative strength. Therefore, the panel design ultimate allowables have been

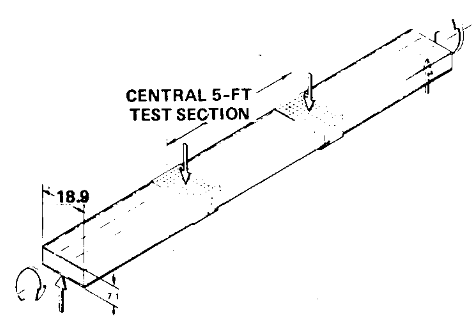

Fig. 3 Large-scale demonstration component.

defined as 0.5 times the parent material ultimate allowables, except in one case (panel 5) where the allowable obtained was directly related to the simulated component.

Honeycomb Sandwich Repaired from Both Sides

The faces of panels 1 and 2 were each 8-ply laminates $(\pm 45/0/90)_s$ with 0.50 in. thick 5056 aluminum honeycomb core, ⅛-in. cell size, 8.1 lb/ft^3. The panel room temperature design ultimate allowables are: tension 1715 lb/in. per face, compression 1900 lb/in. per face, and shear 709 lb/in. per face based on the data in Table 1 and using the factor of ½ for fastener holes and ½ considering the number of plies.

An oval hole 6.6×12.0 in. completely through panel 1 was repaired with access to both sides. The panel faces were scarfed adjacent to the hole on 36/1 slopes using a portable drum sander. The core was replaced and spliced. Then each face was repaired with a bonded and cocured patch that replaced the removed material $(\pm 45/0/90)_s$ and added five additional plies $(\pm 45/0/\pm 45)$. The patch was designed to restore axial load and shear capability. The completed repair is shown in Fig. 4.

The panel was initially loaded in tension by applying bending to the box beam. Loading was terminated at 106% of the tension design ultimate allowable when the maximum measured strain reached 7333 μin./in. on the centerline of the panel about 6 in. off the patch. The loading jacks were reversed and the panel was loaded in compression to 100% of the panel compression allowable when the maximum

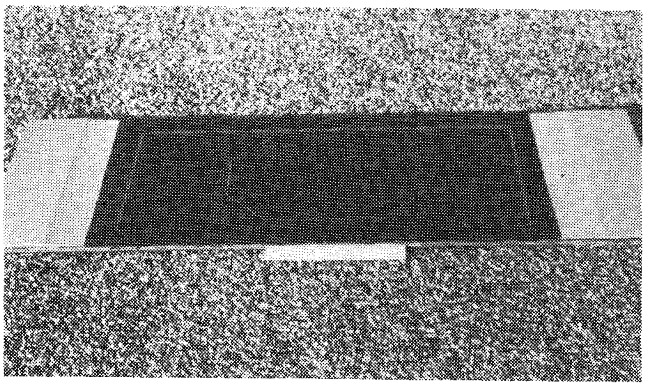

Fig. 4 Repaired panel 1, external surface.

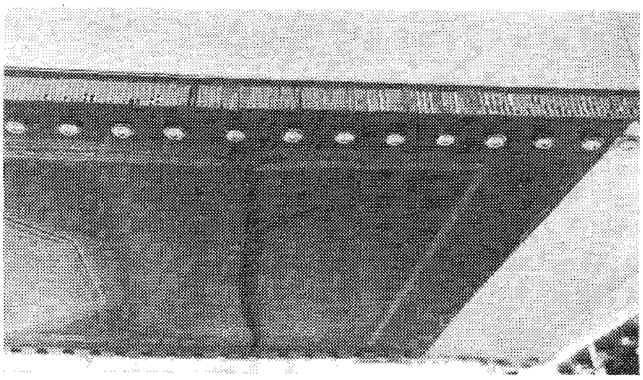

Fig. 5 Panel 1 tension failure at 139% of design ultimate allowable.

measured strain reached 7333 μin./in. Torsion jacks were installed to load the panel in shear. Loading was terminated at 131% of the panel shear allowable with the maximum measured shearing strain at 8422 μin./in. on centerline gages near the patch. The panel showed no indication of failure during and at the completion of these tests.

Bending was again applied to the box beam until tension failure of the panel occurred. Failure loading in the 8-ply outer face occurred at 139% of the panel design ultimate tension allowable. The maximum axial strain at failure was 9670 μin./in. The failure location is remote from the patch, as shown in Fig. 5, initiating at a fastener hole where the measured strain level was 7000 μin./in. away from the local concentration at the edge of the hole.

Honeycomb Sandwich Repaired from One Side

Panel 2 was identical to panel 1 except for the repair of the inner face. The penetration of the inner face was a longitudinal hole 1.0×6.4 in. The outer face and core were removed to a 6.6×12.0-in. hole (as on panel 1). The inner face was scarfed with the drum sander guided by the hole in the outer face, as shown in Fig. 6. The outer face was scarfed with the drum sander guided by an oval template, completing the preparation of the panel for repair.

The patch for the inner face consists of two precured details and one cocured detail. A five-ply precured laminate was made to fit against the blind side of the inner face. A drill template was used to assembly drill the inner face and the blind side patch with a series of small holes (0.070-in. diam, 49 places). With all holes complete, FM-400 adhesive was applied to the patch and the patch was then installed using a length of wire through matching holes.

Once the patch was located the drill template was repositioned over the holes using the wire to hold the patch. Clecos were then installed in all holes, as shown in Fig. 7, to provide pressure during the bonding of the blind side patch. The clecos had been coated with Freecoat 33 and baked at

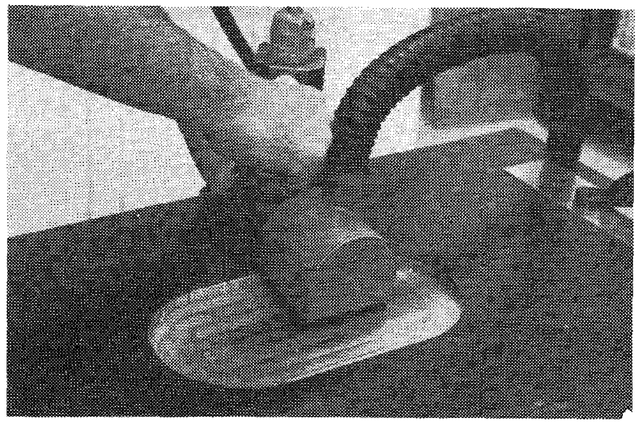

Fig. 6 Scarfing the inner face of panel 2.

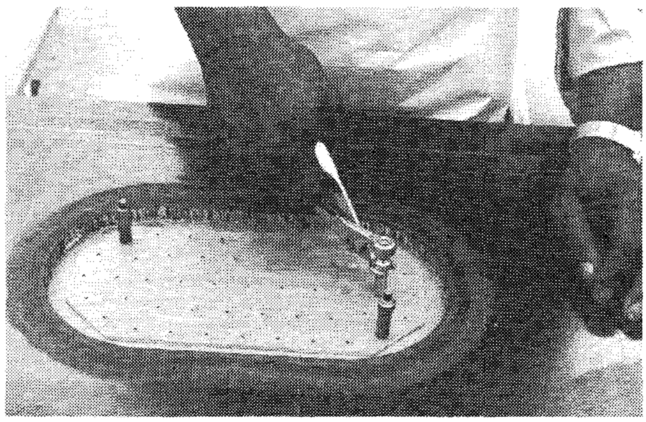

Fig. 7 Installation of clecos for bonding of blind side doubler.

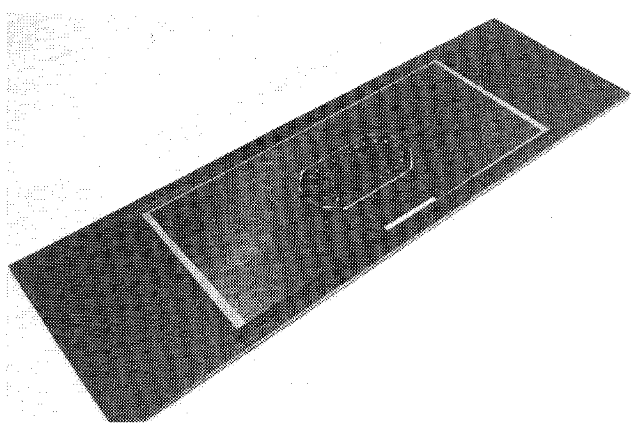

Fig. 8 Repaired panel 2, inner surface.

350°F for 1 h. After curing of the blind side patch adhesive, the clecos were removed and the small holes sealed with adhesive.

With the damage hole now closed, the near side parts of the inner face were installed and vacuum pressure cured. The cocured detail, the replacement material, was installed over film adhesive against the blind side patch and the scarf. Then over an additional film of adhesive another five-ply ($\pm 45/0/\pm 45$) precured detail was installed and the replacement core was placed over the patch. The vacuum bag was laid over the core and sealed to the outer surface. When the inner face repair was complete and inspected the core plug was reinstalled with film adhesive against the inner face and FM-404 forming adhesive in the core splice.

The final operation consisted of cocuring the outer face patch plies with film adhesive over the core, the scarf, and the outer surface. The completed repair as seen from the inside is

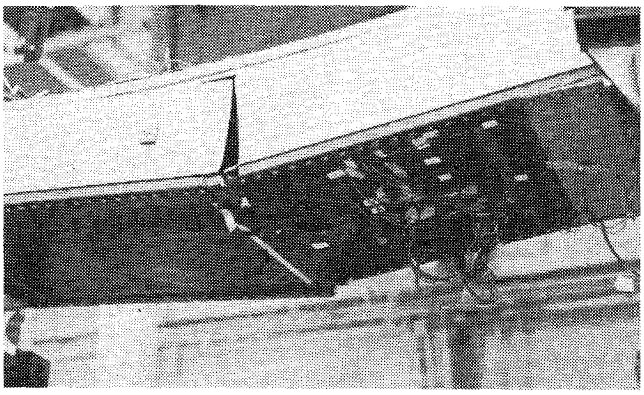

Fig. 9 Panel 2 failure of 122% of design ultimate allowable.

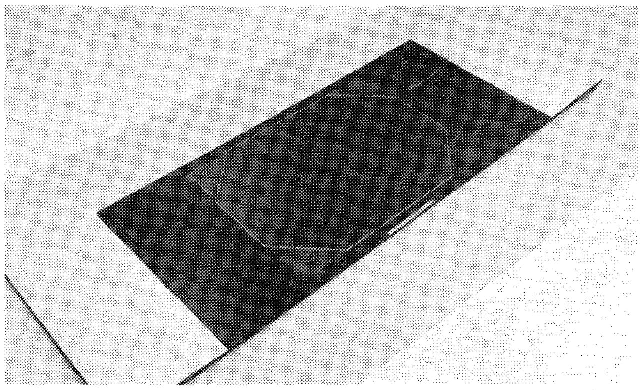

Fig. 10 Panel 3 completed partial thickness repair.

Table 2 Limit loads for panels 3 and 4, B-1 vertical stabilizer skin

Limit load condition	N_X, lb/in.	N_Y, lb/in.	N_{XY}, lb/in.
1) Max. pos bending	3503	83.0	546.0
2) Max. neg bending	3503	83.0	546.0
3) Max. torsion	2380	49.0	1027.0

shown in Fig. 8. The edges of the blind side patch detail were imperfectly bonded at best. Inspection of this patch from the inside is a luxury not available to an airplane repairman. The repair was therefore tested as is with no attempt to improve it.

Panel 2 was loaded to failure in tension at 122% of design ultimate allowable. Maximum strain at failure was 8925 μin./in. on the parent laminate just off the end of the patch. Noises and strain gage readings during loading indicate some deterioration of the bond of the blind side patch. However, the failure occurred outside this area and the bond deterioration appears to have had no effect on the ultimate failure. A photo of the failure, viewed from the outside is shown in Fig. 9. The test results and the mode of failure demonstrate that even an imperfectly bonded repair, made simulating on-aircraft conditions with access from one side only, can restore strength well beyond ultimate design load requirements. By using a repair method that is capable of high strength restoration, a considerable margin is available to allow for reduced quality standards.

50-Ply Laminate Partial Thickness Repair

The panel 3 laminate $[\pm 45/90_2/\mp 45/\pm 45/0_4/\mp 45/\pm 45/0_3/\mp 45/\pm 45/0_2]_s$, duplicates a section of the vertical stabilizer skin of the B-1 airplane. A parent laminate tension ultimate allowable of 18,270 lb/in. at room temperature was determined from the results of six tests.[4] The skin is obviously stiffness designed, considering the maximum limit loads shown in Table 2.

Panel 3 represents the repair of an extensive surface damage partially through the laminate. All damaged material was removed and the hole dressed to the desired shape. Material was removed from the outer surface of panel 3 to a maximum depth of 0.120 in. (21 plies) throughout an 8 × 12-in. oval. The edge of the cutout area was then scarfed to a larger 12 × 20.4-in. oval using the drum sander with a variable angle track. Experience has shown that a skilled operator can cut such a scarf with various tools. The track is a practical way of giving some control to guide the cutting tool. The variable scarf angle minimized the width of the patch and the amount of material removed while yet providing for a sufficiently long scarf in the critical areas. The end scarf slope is 36/1 and the side slope is 18/1.

Pursuing the principal objective of restoring the original strength and stiffness of the panel, a 24-ply cocured patch was installed with FM-400 adhesive against the scarf and a lightly

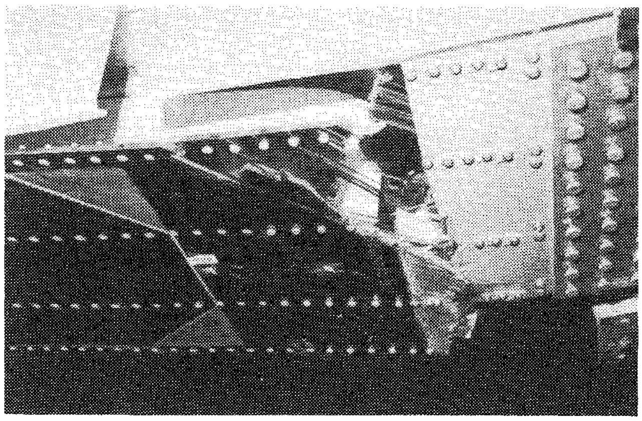

Fig. 11 Panel 3 failure at 155% of design ultimate allowable.

sanded outer surface. The patch laminate $[\pm 45/0_3/\mp 45/0_5/\mp 45/90/\mp 45/\pm 45/90/(\mp 45)_2]$, listed from inside outward, replaced all cut plies plus one extra ply in the 0, +45, and −45 deg directions. The order was slightly changed from the parent only so that the 0-deg groups would be attached more closely to those groups in the parent laminate. The completed repair, oven cured under vacuum pressure at 350°F, is shown in Fig. 10.

Testing of panel 3 proceeded first by applying the limit loads of Table 2 (neglecting N_y) with no indication of failure. Load condition four was positive bending to failure, which occurred at an average tension panel loading of $N_x = 14{,}200$ lb/in. The maximum axial strain recorded in the panel and projected to failure was 7720 μin./in. Failure occurred remote from the repair, as shown in Fig. 11. Again using the 50% strength reduction factor for the skin/spar attachment holes the panel ultimate allowable would be 9135 lb/in. Hence, the repair sustained 155% of the panel design ultimate allowable strength.

50-Ply Laminate Accessible from Both Sides

Panel 4 represents the repair of a complete penetration damage of a highly loaded laminate. The laminate was identical to panel 3 except for the repair. A 4-in.-diam hole was bored through the panel and the hole scarfed from both sides at a slope $L/t = 36/1$. The scarf was 27 plies deep from the outside, 23 plies deep from the inside. A patch of 61 plies (32 outside, 29 inside) was cocured and bonded with FM-400 adhesive under vacuum pressure at 350°F. Of the 11 extra plies, 7 were at 0-deg orientation and 4 were at ±45 deg, all interspersed among the 50 replacement plies.

The testing of panel 4 proceeded in the same manner as for panel 3. No indication of failure was seen from the three limit load conditions. Positive bending to failure resulted in an average panel loading of $N_x = 14{,}200$ lb/in., almost identical to the load required to fail panel 3. The maximum strain at

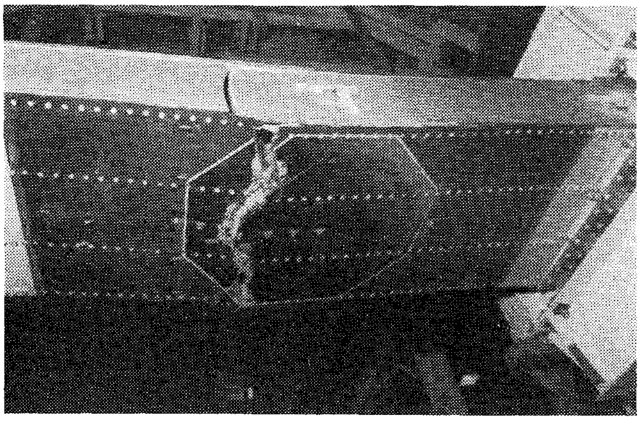

Fig. 12 Panel 4 failure at 155% of design ultimate allowable.

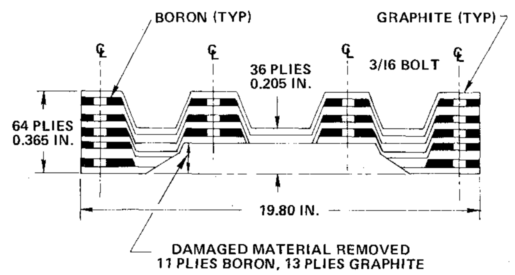

Fig. 13 Cross section of panel 5 as damaged.

failure was 6920, μin./in. with a more uniform distribution than in panel 3. Failure occurred through the repair, as shown in Fig. 12. In spite of some irregularity in this repair the results were quite satisfactory.

64-Ply Boron-Graphite/Epoxy Hybrid Laminate

Panel 5 was an exact representation of a section of the horizontal stabilizer skin of the B-1 airplane with local buildup over spar caps using unidirectional boron/epoxy strips. The typical cross section is seen in Fig. 13. The damage extended 24 plies deep removing 11 of the 28 plies of boron over the two interior spar caps plus all the associated graphite between the spar caps. The scarf in this section involved two slopes to minimize cutting of shear plies in the panel. Ideally, only 13 plies of graphite were to be cut by the transverse scarf on a slope of 0.10 in. per ply. The longitudinal scarf length is 7.20 in. or 0.30 in. per ply. The panel after material removal is shown in Fig. 14.

The embedded boron fibers in this panel complicated the material removal significantly. A diamond-coated end grinder was used to make the 9.44 -in.-diam 24-ply damage cutout and a diamond-coated drum grinder was used on the scarf surface. A metal track was made to guide the drum grinder.

At the completion of the material removal machining, the inclined surface of the scarf was examined to locate the end of each parent laminate ply and to compare its location with the theoretical location shown on the repair drawing. Because of minor variations in cured ply thicknesses in the parent laminate, the locations of ply ends at the scarfed surface varied from their theoretical location by amounts up to 0.3 in. The size and shape of the patch was modified as required to accommodate these variations. By placing a semitransparent sheet of mylar film over the scarfed surface, the ends of each ply were noted, the desired overlap was allowed, and a series of patterns was drawn on the mylar, one for each ply. By laying the mylar film over the prepreg and cutting along the lines, starting with the largest piece first and working in toward the smallest, each ply was cut to the desired size. The plies were laid up on a paper print made from the mylar, starting with the smallest and working out to the largest, so

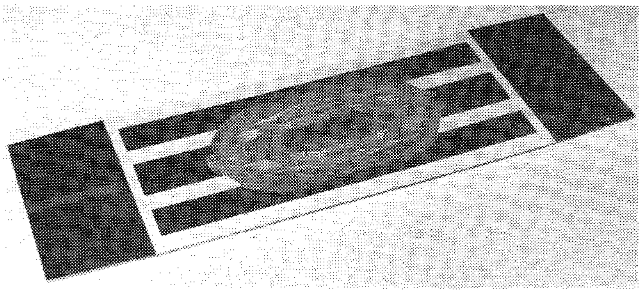

Fig. 14 Panel 5 after material removed.

Table 3 Limit loads for panel 5, B-1 horizontal stabilizer covers

Limit load condition	ϵ_X, μin./in.	N_{XY}, lb/in.
1) Max. tension with shear	2023	725
2) Max. compression with shear	−2023	−725
3) Max. shear with compression	−1469	841
4) Max. shear with tension	1469	841

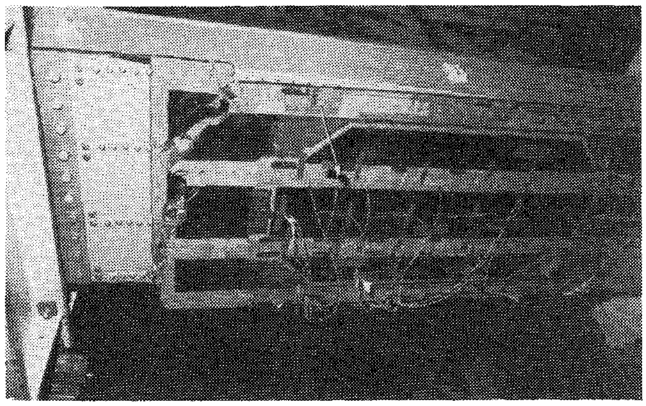

Fig. 15 Panel 5 failure at 151% of design ultimate allowable.

that the pattern of the print located each ply. The resulting book of patch plies was then laid into the machined area of the parent laminate over a layer of RB-398 adhesive. The procedure as described is recommended for use on actual repairs since it is a reasonably fast method of obtaining patterns and results in patch plies which match the actual geometry of the machined surface.

Strips were cut from a sheet of flame-sprayed aluminum approximately 0.010 in. thick to replace the lightning protection system in the repair area. After all the patch plies had been placed, the replacement lightning protection strips were laid over the patch plies. No adhesive was used under the aluminum strips where they were in contact with uncured prepreg of the patch plies. At the ends of the strips, a small length of RB-398 film adhesive was placed between the parent laminate and the aluminum strip. The entire assembly was bagged and cured.

The bond between the aluminum and the graphite was intact after curing for the full length of both aluminum strips. However, some wrinkling of the aluminum strips occurred at two locations on each strip. This wrinkling appears to be the result of differential thermal expansion during curing, and suggests that a secondary bond using room temperature cure would be preferable. The wrinkles distorted the underlying graphite plies slightly, but since the distorted plies near the surface are either 45 or 90 deg orientation, the distortion was not expected to seriously affect the strength in the 0 deg primary load direction.

The four limit load conditions of Table 3 were applied to the panel followed by four ultimate load conditions.[5] No indication of failure was observed.

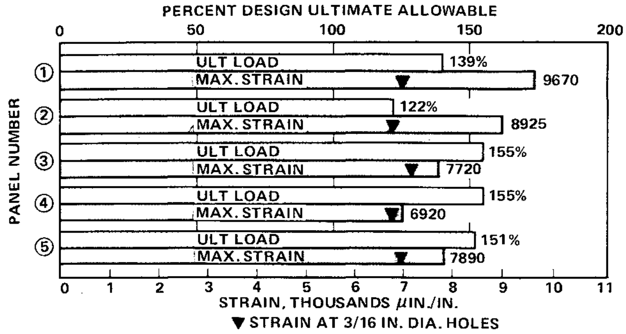

Fig. 16 Summary of demonstration panel results.

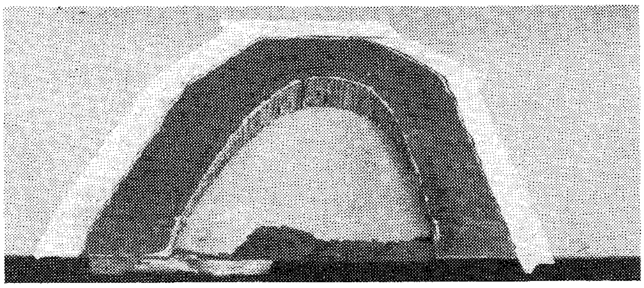

Fig. 17 Repair area of F-15 speedbrake.

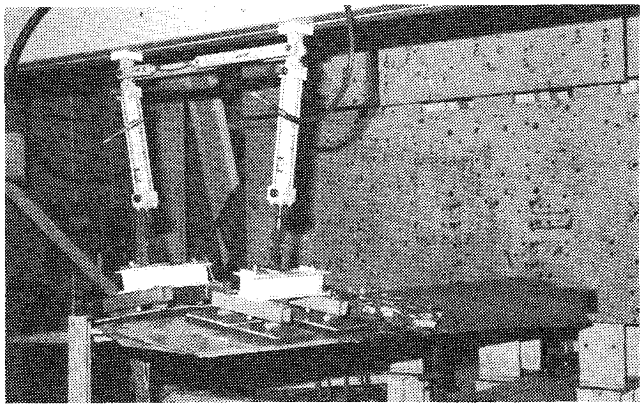

Fig. 18 F-15 speedbrake test arrangement.

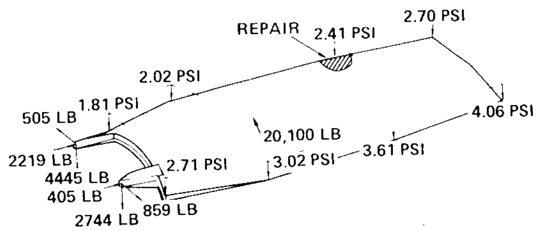

Fig. 19 F-15 speedbrake proof test limit load case.

The panel design ultimate allowable was defined by Grumman Aerospace[6] as that load resulting in a strain of 4171 μin./in. The failure load condition imposed uniform bending on the box resulting in tension in the repaired panel. Failure occurred at a panel load intensity of 19,300 lb/in. at a location remote from the repair as shown in Fig. 15. On a typical section outside the repair the mean measured strain was 6300 μin./in. Hence the panel failure occurred at 151% of design ultimate load. The maximum parent laminate strain was 7912 μin./in. The maximum strain level near a fastener hole was estimated 6900 μin./in.

Summary of Test Panel Results

A summary of the test results of panels 1-5 is shown in Fig. 16 giving the percent of design ultimate allowable and the maximum strain at failure. The mark on the strain bar indicates the strain extrapolated or interpolated to the spar attachment hole centerline not including the local effect of the holes. Since each failure apparently originated at a hole, this strain appears to be the best correlating parameter. In any case, the failure loads, strains, and failure modes of these five repaired panels demonstrated that it is possible to restore a badly damaged mechanically attached graphite structure to its original strength which is well above current design ultimate allowables.

F-15 Speedbrake Repair

Several speedbrakes for F-15 aircraft were fabricated with T300/5208 graphite/epoxy skins.[6] The speedbrake is approximately 10-ft long and 3½-ft wide and contains machined titanium actuator fittings and hinge arms which are integrally bonded with the graphite/epoxy skins and aluminum honeycomb core. Damage on one speedbrake consisted of an indentation on the outer face approximately 6×7 in. in area and 0.25 in. deep, located near the edge of the part about 3 ft forward at the aft end. Fibers were not broken, but the edge was split, the inner surface was bulged slightly, and radiographic inspection showed the core to be locally crushed.

All removal of damaged material was done with hand tools and no special guides to demonstrate a simple technique that could be used for on-aircraft repairs. A hand held router was used to cut the 5-6-ply skin. The cut was made to a scribed line, the skin was peeled off the core, and the core cut with a knife.

At the edge of the opening that had been cut in the outer skin, the skin was scarfed over a width of approximately an inch using a small belt sander. The scarfed surface was cut to end at a scribed line on the surface. The repair area with the core removed and the scarf surface prepared is shown in Fig. 17.

A heat blanket was used for curing the repair on the F-15 speedbrake. A core patch was bonded in place using FM-404 foaming adhesive for the core splice, FM-400 adhesive for the inner skin-to-core bond, and Proseal 828 as the core edge potting compound. The heat was controlled by monitoring a thermocouple on the laminate and manually adjusting the current flow to the blanket.

As a separate, second curing operation, the repair of the outer skin was made using the typical configuration developed earlier in the program, i.e., a scarfed surface of the parent laminate, a layer of FM-400 film adhesive, 5 plies of AS/3501-6 unidirectional tape prepreg replacing parent laminate plies (plus a 0-deg ply at the inboard side of the patch where the parent laminate contained 6 plies) and 2 added plies on the surface extending beyond the end of the replacement plies and bonded to the surface of the parent laminate. Serrations were also used at the ends of outer plies. Paint was removed in the repair area by light hand sanding with fine sandpaper. An 0.020-in. thick aluminum pressure plate was used to improve the uniformity of heating. The blanket was placed on top of the nylon bag and separated from it by four layers of cloth. Four more layers and additional insulation were placed over the blanket.

A small area of the inner skin approximately 3.0×1.5 in. in area was repaired similarly to the outer skin. In order to obtain a near-flush surface, two plies of E-glass epoxy were placed against the core as a filler prior to making the graphite/epoxy skin patch. A small blister occurred on the inner skin patch due to unknown causes. The void was discovered by visual and coin tape examination and was injected with Epon 815 epoxy. Radiographic inspection made after the repair was completed, indicated no significant defects in the replacement core or skin patches.

To verify the adequacy of the repair, the speed brake was proof loaded as shown in Fig. 18. The unsymmetrical loading condition shown in Fig. 19 was applied, imposing only those pressures aft of the repair with loading pads in compression.

Fifteen strain gages were used to monitor the test loading. The peak strain recorded was 5583 μin./in. adjacent to the patch at 120% of the limit load. No indication of failure was observed.

Conclusions

1) Advanced composite structures can be repaired to restore between 80 and 100% of the parent laminate ultimate allowable. This level of strength restoration is approximately twice current design allowables.

2) Northrop bonded-cocured scarf repairs have been shown to be permanent, durable, and not unduly affected by adverse environmental conditions.

3) No size limit is imposed by the design approach described here.

4) No expensive special tools are required for advanced composite repair.

5) Personnel training will be required for these techniques which have the potential of high quality depot repairs and of fast/adequate field repairs.

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