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Composite Material Repairs to Metallic Airframe Components

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The repair of metal structures by using composite materials is a technology that was pioneered in Australia, and excellent results have been obtained in over 500 repairs of corrosion and fatigue cracking. This paper will present details of the first in-service composite material repair applied to metal structure on USAF C-141B operational aircraft. This engineering study evaluated five locations representing both fatigue and corrosion cracking susceptible areas on the C-141B strategic airlift transport aircraft. Repairs using either boron/epoxy or graphite/epoxy are beneficial for crack retardation, end fastener effect relief, and field stress reduction. The most obvious time saving feature of these repairs is that no fastener removal or installation is involved. Extensive structural analysis utilizing finite element modeling was done to ensure that the repair design not only would have the requisite strength and durability but also would reduce the inspection burden after the aircraft was returned to service. A test program was also undertaken to substantiate the analytical investigation. Typical results from the finite element models as well as the specimen and component tests will be presented. Finally, the step-by-step procedure by which an actual boron/epoxy repair was applied to a fatigue crack on a C-141 center wing will be reported and "lessons learned" will summarize the benefits of applying the technology.

Introduction

IT is, and has been for many years, common practice to repair aircraft structures using repair doublers of like material and to attach to the structure with metal fasteners. This approach has the advantage of compatibility of materials and simplicity of analysis. In cases where the repair member thickness is restricted, however, a material with strength and modulus of elasticity greater than the damaged material should be used. Also, when a repair doubler is installed with fasteners, additional fatigue crack sites are created at the fastener holes, where eventual cracks may result. Repairs of metal structures using bonded doublers made from composite materials offer the potential of increased time between inspections and reduced repair time and provide the advantage of thinner repair members and elimination of repair fasteners.

The repair of metal structures by utilizing composite materials is a technology pioneered in Australia by Dr. Alan Baker for use on Royal Australian Air Force (RAAF) aircraft. Dr. Baker and his associates at the Aeronautical Research Laboratories (ARL) in Melbourne, Australia have repaired C-130 (Hercules), Mirage, F-111, Macchi, and P-3 (Orion) aircraft using Carbon Fiber Reinforced Plastic (CFRP) and Boron Fiber Reinforced Plastic (BFRP) repairs. Over 500 repairs have been made with excellent operational service experience. Dr. Ratwani and his colleagues at the Northrop Aircraft Co. have also been very active in the study and testing of such composite material repairs.

Following a presentation in 1981 at the U.S. Naval Research Laboratories by Dr. Baker, the System Program Management Division at the Warner Robins Air Logistics Center (WR-ALC) Robins AFB GA, submitted a logistics need to the Air Force Coordinating Office for Logistic Research (AFCOLR) Wright-Patterson AFB OH, requesting that research be undertaken on this topic with applications to USAF Transport aircraft. This prompted work by both the Air Force Flight Dynamics Laboratory and the Lockheed Aeronautical Systems Company-Georgia (LASC-GA). Lockheed began studying the concept of using composite materials for the repair of C-141 airframe components in 1984. Studies were conducted under Independent Research and Development (IRAD) projects which lead to WR-ALC awarding LASC-GA a contract to study this concept for use on the C-141B weapon system. This paper covers the selection of candidate locations, selection of configuration and materials for repairs, and the analytical evaluation of the repairs for structural adequacy and durability on the C-141B.

1) The areas selected for repair are actual C-141 potential problem areas or are based on damage found on the C-141A fatigue test article and in-service aircraft.

2) The repair is to be bonded to the structure, and fasteners are not to be used for its attachment to structure. This is done to eliminate additional fatigue sites and need for inspections.

3) The structure to be repaired is to be distributed as little as possible. This means that fasteners in the repair area are not to be removed unless necessary to inspect or ream the hole for increased service life. This is done to eliminate the possibility of damage due to removal and installation of fasteners.

4) Maximum advantage is to be taken of the use of available materials, fabrication and installation methods, test data, and service experience. This use of proven materials and methods reduces the risks, development time, and costs. This is accomplished by literature survey and visits and coordination with organizations having previously successful developments, such as ARL.

5) In keeping with the goal of minimum development testing, detailed Finite Element Models (FEM) are used extensively to predict the stress distribution in the original structure. The detailed FEMS are also used to determine the relative merits of the various repair configurations in reducing stress levels.

6) The repair is sized such that it reduces the stress level in the repair/prevent area by a minimum of 15%. This reduced

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stress level will result in significant retardation in crack growth rate as well as reduced inspection burden, down time, and costs.

7) The repair members are fabricated with balanced plies to minimize warping during curing.

8) The doublers are gradually tapered at the ends to reduce peeling stresses and to minimize centroidal changes.

Design Studies for Repair Locations

The design of composite repairs for primary structural areas of the C-141B aircraft is based on two primary criteria. First, the general repair technology is based on previous work for ARL and the RAAF. This significantly reduces cost, time, and risk by utilizing a proven technology. Second, a research and development phase will be undertaken only to make the optimum use of this technology for the selected structural areas on the C-141 aircraft. These composite repairs will form the basis from which future repairs or preventive maintenance actions can be implemented on in-service aircraft. Each repair area design will follow the problem identification, preliminary solution, refinement, analysis, and decision stages before final engineering drawings are released for trial installations.

This feasibility study tasked by WR-ALC resulted in selection of five locations on the C-141B aircraft for study of repair/reinforcement with composite materials. The locations are identified as: 1) inner wing to outer wing lower surface rear beam joint—WS 405; 2) wing lower surface panel riser weep holes and rib clip attachment holes—typical on inner wing; 3) WS 77 inner wing aft corner fitting to beam cap; 4) vertical stabilizer/dorsal longeron intersection at vertical stabilizer front beam; and 5) FS 998 fuselage/main landing gear frame.

These five areas were chosen on the basis of need and not necessarily for ease of application of composite repairs. The areas chosen provide a good cross-section of problems facing the composite repairs. These repair locations are in Fig. 1. Selection of the five study areas is based on cracks found on the C-141A fatigue test article and on in-service aircraft. As the C-141B force exceeds 30,000 flight-hours, the probability of cracking in these areas increases. The extensive costs and down time involved with conventional metallic repairs makes composite repair an attractive alternative to a metal design for repair or as a reinforcement to improve fatigue endurance and stress corrosion resistance.

Inner Wing to Outer Wing Lower Surface Rear Beam Joint—WS 405

The WS 405 joint is composed of a large 7075-T6 aluminum alloy fitting which splices the rear beams of the inner and outer wings. This fitting splices the wing lower surface panels, rear beam caps, and webs. It also provides an attachment to the Inner Wing Box Rib (IWBR) Station 374 rib. A jacking pad is incorporated in the lower portion of the fitting. Thus, the fitting provides structural continuity while accommodating the different airfoils of the inner and outer wings and provides a jack pad and tiedown fitting.

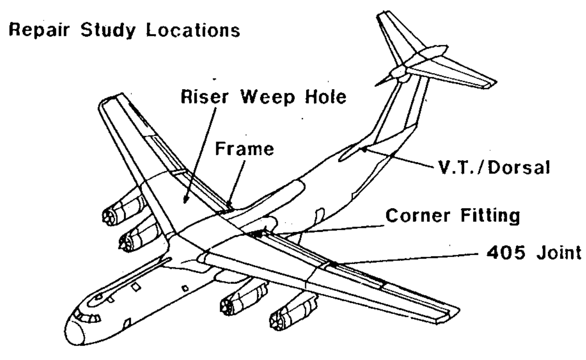

Fig. 1 C-141 repair using composite materials.

The complex geometry and surrounding structure are major constraints in the repair of the lower WS 405 area. In addition, the aircraft must retain its present tiedown and jacking capability in this area. The repair of a damaged WS 405 splice fitting to restore the capability to transfer loads from the outer wing beam cap, wing panel, and web to the inner wing is a complex matter, so several repair configurations were evaluated. The configuration finally selected was an external doubler bonded to the inner and outer wing panels, located below the beam caps. Since no fasteners or sharp radii are involved, the advantage of higher stiffness of boron/epoxy material can be taken. The doubler will not cover the tie down fitting and will permit standard mooring of the aircraft. Due to the vertical jacking load imposed on the WS 405 fitting, a metallic insert is required, as the epoxy matrix will not be able to withstand these high concentrated loads. A load carrying metallic insert having composite legs was considered, but splicing it effectively to the composite doubler proved too difficult. The resulting composite requires 48 plies of unidirectional AVCO 5505/4 boron/epoxy tape with a titanium insert for the WS 405 jacking loads. Each leg is parallel with its respective rear beam datum, forming a "bow tie" shape. The general shape is shown in Fig. 2.

The doubler is bonded using an American Cyanamid FM 73 film adhesive. Ultrasonic inspections are performed to check for voids in the doubler and in the adhesive bond line. The doubler is then covered with one layer of aluminum foil and sealed around the edges. This provides environmental protection by keeping moisture away from the doubler and the adhesive. The rework area is then painted to match the aircraft. The WS 405 doubler was the first part to be designed, fabricated, and installed. Actual WS 405 joint area was not available on the static test article (Aircraft S/N 6000) for trial installation, so a similar compound contour area on a pylon was used. The fabrication and installation experience gained from this location was applied in the repair design of the other areas.

Wing Lower Surface Panel Riser Weep Holes

The weep holes are nominally 0.25-in.-diam. holes drilled through the panel risers and approximately 0.25 in. above the inner surface of the wing panel. They help increase usable fuel by allowing fuel to flow between the riser bays. Approximately 30 weep holes are in each inner wing lower surface riser. The stress concentration caused by the weep holes adversely affect the aircraft service life.

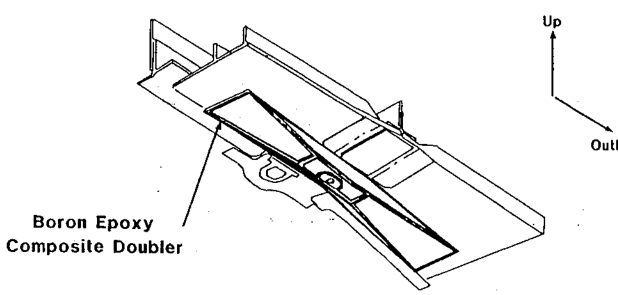

Fig. 2 C-141 WS 405 rear beam splice composite repair.

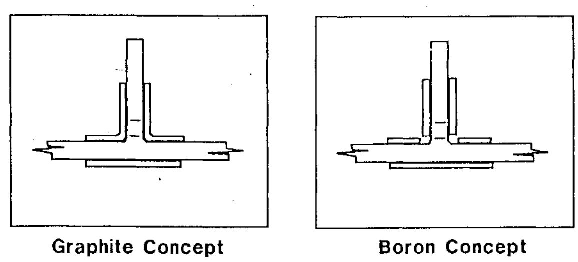

Fig. 3 C-141 weep hole composite repair concept.

The repair concepts for the weep hole area included back-to-back composite angles nested on the risers and the adjacent panel surfaces. An additional doubler was necessary on the exterior of the lower surface to minimize the eccentricity and to retard the growth of crack propagating down in the panel. Although this is a structurally viable solution, manufacturing considerations eliminated it. The riser fillet radius varies from 0.06–0.09 in. However, the large fiber stiffness of boron filaments limits it to a 0.50-in. minimum bend radius and in case of graphite filaments to a minimum 0.12-in. bend radius. Even if a graphite/epoxy material could be bonded around the radius, there is high probability of disbond in the radius area.

The optimum solution was to use a five-piece boron/epoxy doubler arrangement. The internal doublers are placed as close to the radius as possible, as shown in Fig. 3. Since the surfaces are basically flat, the boron/epoxy tape can be cured at 350°F, utilizing an autoclave process. The material is an AVCO 5505/4 boron/epoxy prepreg tape. Each piece is 6 in. long and consists of 14 plies (total thickness 0.074 in.). The cracked weep hole is fitted with an aluminum plug to prevent future deformation of the hole under applied loading. At weep hole locations where 0.020-in.-high by 1.0-in.-wide raised pad exists, additional plies of the film adhesive can be used as shims. After bonding and ultrasonic inspection for voids, the internal doublers are overcoated with sealant. The internal doublers will receive a coat of flexible polyurethane, and the outside doublers will receive the aluminum foil/paint treatment.

WS 77 Inner Wing Aft Corner Fitting to Beam Cap

The wing station (WS)77 inner wing lower corner fitting is a large aluminum fitting inside the inner wing at the rear beam WS 77.7/FS 958 intersection. This fitting transfers the inner wing beam cap loads to the center wing beam caps via large tension bolts. The problem caused by this fitting results from the abrupt drop off in thickness at the outboard end of the horizontal leg. This abrupt ending of the fitting causes large shifts in the load path and induces high bending stresses (i.e., end fastener effect) in the rear beam caps. The fitting is shown in Fig. 4.

Internal composite members were first considered for the WS 77 rear beam corner fitting area. A graphite/epoxy molded part would nest on the vertical and horizontal legs of the corner fitting and provide a gradual transfer of loads in the beam cap. The lighter weight is the only advantage this part would have over a metallic part. Fasteners would still be required in the fitting due to the high load transfer. In addition, the mating of several different mating surfaces would cause fit problems which could lead to failure of the adhesive.

Finally, the simple and easy method, consisting of bonding an external composite doubler to the lower wing surface was chosen. This would provide the loads in the horizontal leg of the beam cap an alternate load path, thus lowering the beam cap stress levels and retarding the onset of cracking. The final design consists of a rectangular doubler fabricated from 32 0 piles of AVCO 5521/4 250°F curing boron/epoxy tape. A gentle taper is provided in the inboard-outboard direction. The doubler is cured and bonded at the same time to the wing panel. Based on ARL experience, the laminate made from all 0-ply orientations is used. Since all plies are of the same orientation, ply drop off about the center line is not necessary to maintain symmetry. After bonding, the composite doubler is inspected for voids, covered with aluminum foil, sealed, and painted.

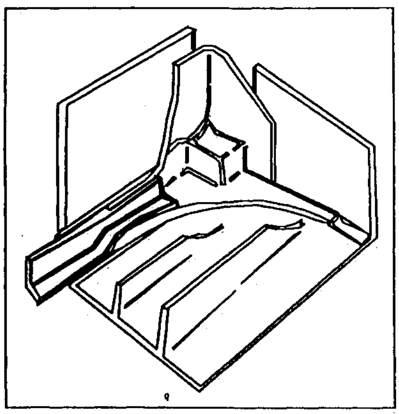

Fig. 4 C-141 corner fitting.

Vertical Stabilizer/Dorsal Longeron Intersection at Vertical Stabilizer Front Beam

The low-level missions usage of C-141B aircraft imposes higher stresses than originally anticipated on the dorsal longeron and vertical stabilizer front spar area. This fact and the complex contour of the area led to the selection of this area for development of a composite repair. The basic structure is shown in Fig. 5. The dorsal longeron, located near the crown of the fuselage, is a large "T" shaped extrusion and intersects the center spar of the vertical stabilizer. The vertical stabilizer front spar consists primarily of two aluminum extruded "T"-shaped spar caps, one on the left and right sides with a connecting web. The front spar is discontinuous at the dorsal longeron and is spliced through by bath tub fittings and tension bolts. In-service aircraft have experienced fatigue failures of the front spar splice fittings, splice plates, and spar caps.

The complex internal geometry and the locations of existing fasteners dictate the use of external repair members as the only practical solution. The splice area of the front spar of the vertical stabilizer requires stress level reductions, so the doubler is primarily sized for spar loads. Detailed FEM analysis showed that a rectangular doubler was adequate, which greatly simplified the fabrication of the part.

The doubler is approximately 30 in. long, 0.168 in. thick, and composed of 32 plies of boron tape. The AVCO 5521/4 tape cures at 250°F, so the compound contour can be permanently shaped into the doubler at the same time it is being bonded in place. The same doubler can be used on either the left or the right side of the front spar. The tapers on the top and bottom of the doubler are different due to the stiffness differences of the front spar cap above and below the dorsal longeron. After bonding, the doubler is checked for voids, covered with aluminum foil, sealed, and painted to match the aircraft.

FS 998 Fuselage/Main Landing Gear Frame

The Fuselage Station (FS) 998 Main Frame is a heavy frame member machined from a large 7075 aluminum alloy forging and heat-treated to the T6 temper. The frame is 40 in. aft of the center wing rear beam and is also used as the main landing gear support member. The frame has a large hub for the landing gear interface. The outboard cap of the frame follows the contour of the fuselage. Inboard and outboard caps are

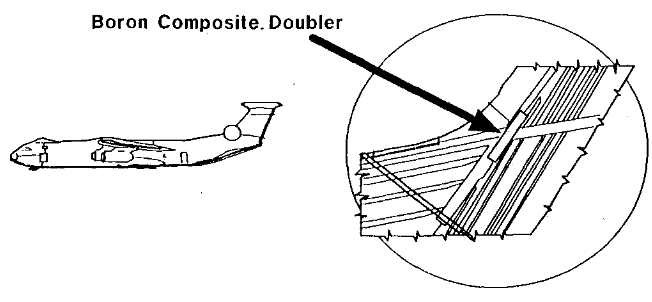

Fig. 5 C-141 dorsal/vertical stabilizer composite repair.

connected by an integral web and supported by integral stiffening members, forming many "pockets" in the frame. Figure 6 shows the frame. The primary mode of cracking experienced in service aircraft is stress corrosion and is in the pocket area. This stress corrosion is due to the residual stresses caused by the extensive machining as well as the low stress corrosion threshold level of the 7075-T6 forged material.

The repair concept consists of two graphite/epoxy angles positioned on each side of the frame. Graphite/epoxy is required due to the 0.50-in. fillet radius.

To keep the stress corrosion crack from growing, a good bond between the radius and composite repair angle is necessary. Load transfer and prevention of moisture are the other considerations that were addressed. The repair angle is sized for a 2-in.-long crack, as being likely to be found on an operational aircraft.

The angle is approximately 7.5 in. long, 3.5 in. high, 2.7 in. wide, and 0.170 in. thick. It consists of 34 plies of 250°F curing SP-377 (3M) graphite/epoxy tape. The plies are dropped in sets of four about the center line of the laminate. One layer of dry type 120 fiberglass is used as a galvanic barrier between the aluminum frame and graphite/epoxy angle, with a 0.5-in. overlap of the frame. This composite repair member also has rounded corners. It was found that rounding the corner helps to reduce the peeling stresses at the corners. The angles are co-cured and bonded to the frame. The angles are checked for voids, sealed, and painted.

Materials and Processes Study

In 1976, the Australian Aeronautical Research Laboratories (ARL) conducted initial investigations on the feasibility of using composites for the repair of cracked aircraft structures. Small, constant-stress, cantilever 7075-T6 aluminum alloy fatigue specimens were used to evaluate the performance of four different adhesive systems and their effectiveness on crack propagation. Changes in strain transfer efficiency was the primary evaluation criteria.

Four types of adhesives were evaluated by ARL; 1) ethyl cyanoacrylate (one part liquid), 2) flexible epoxy (two component paste), 3) rigid epoxy (two component paste), and 4) expoxy-nitrile (film adhesive). The critical parameters considered in selection of the initial adhesive systems were: static strength, fatigue strength, and resistance under various environmental conditions such as moisture, aircraft fluids, and temperature.

In this screening program several low-temperature curing adhesives were evaluated by the ARL, but none had adequate fatigue or stress-relaxation properties. Therefore, the choice for their initial repair work was 3M's AF126, and epoxy-nitrile structural film adhesive cured at 120°C (250°F) and 300 kPa (43.5 psi). The 3M AF126 adhesive system used in this early

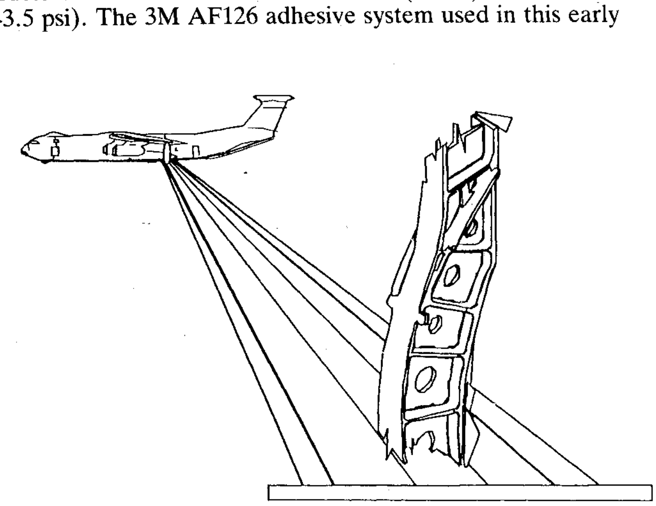

Fig. 6 C-141 FS 998 mainframe composite repair.

work had excellent fatigue and stress-relaxation resistance and adequate bond durability with minimal surface preparation; however, its 120°C (250°F) cure was a major drawback.

Later investigations by the ARL found that FM73M, a structural film adhesive system from American Cyanamid, could be cured at 80°C (176°F) with negligible performance penalty compared to the properties obtained with its recommended 120°C (250°F) cure. FM73M adhesive, a state-of-the-art epoxy nitrile system, although generally lower in performance than AF126, has an improved shelf life and moisture resistance compared to AF126 adhesive. In addition, the FM73M provides excellent durability with simple surface preparation—even when cured at temperatures as low as 80°C (176°F). On the basis of this experience it was decided to use the FM73M.

The ARL preferred boron fiber reinforced plastics (BFRP) over carbon fiber reinforced plastics (CFRP) as a general repair material. Boron fiber offers superior stiffness and fatigue strength and its coefficient of thermal expansion is closer to aluminum than that of carbon fibers. Low electrical conductivity of boron fiber systems allows eddy current inspection equipment to detect minor cracks under repair patches. In addition, boron fiber composite patch forms a barrier to further corrosion. Carbon fibers do form a strong galvanic couple with aluminum, and, therefore, carbon-fiber-based repair doublers require additional corrosion protection measures in the repaired area. However, carbon fiber composites are preferred where the repair must conform to small radii (30 mm or 1.20 in.) or when material cost is a factor. Carbon fiber materials also offer the advantages of ready availability and ease of handling compared to boron fiber composites.

When selecting a boron fiber composite material the basic selection is between two systems supplied by the Speciality Materials Division of AVCO. The two systems, designated 5521/4 and 5505/4, are both epoxy resin matrix materials and differ primarily in cure temperature and elevated temperature performance. For repair work, the AVCO 5521/4 material is preferred because of its lower cure temperature, 250°F vs 350°F for AVCO 5505/4. This also results in lower induced thermal stresses for on-site cured patches and reduced thermal input required to the repair area. The slight penalty in reduced physical properties at lower temperatures (approximately 250°F) has not been a limiting factor in current repair applications.

For graphite/epoxy composite doublers, the choice of material is between basic 350°F cure material such as Hercules 3501-6 or 250°F curing system SP-377 from 3-M Company. The selection of one system over another will be governed by the same reasons discussed for boron/epoxy system.

Surface preparation was also evaluated by the ARL during the initial adhesives investigations. Alumina grit-blasting (with methyl ethyl ketone (MEK) degreasing before and after grit-blasting) was the only surface treatment found effective for both graphite- and boron-reinforced repairs. It was also noted that a simple surface preparation was highly desirable to avoid conditions which would aggravate problems such as stress-corrosion. Therefore, application of chemical surface treatments was minimized during early work to avoid the danger of propagating the pre-existing crack by stress-corrosion and because of the difficulty of performing chemical treatments under field conditions.

Further studies by the ARL had revealed that a silane adhesive promotor, with either alumina grit-blasting alone or grit-blasting with a phosphoric acid nontank anodize (PANTA) served as a simple treatment and provided high bond durability without aggravating the stress-corrosion condition. The adhesive promoter that gave the best results was gamma-glycidoxy-propyltrimethoxy silane (gamma-GPS) sold by Union Carbide as Silane A-187. A typical surface treatment included: abrasion to remove scratches, one MEK wipe, grit-blasting with 50-μ alumina, a PANTA treatment (some cases only), and treatment with a silane solution (1% by volume in distilled water) followed by an air dry prior to application of

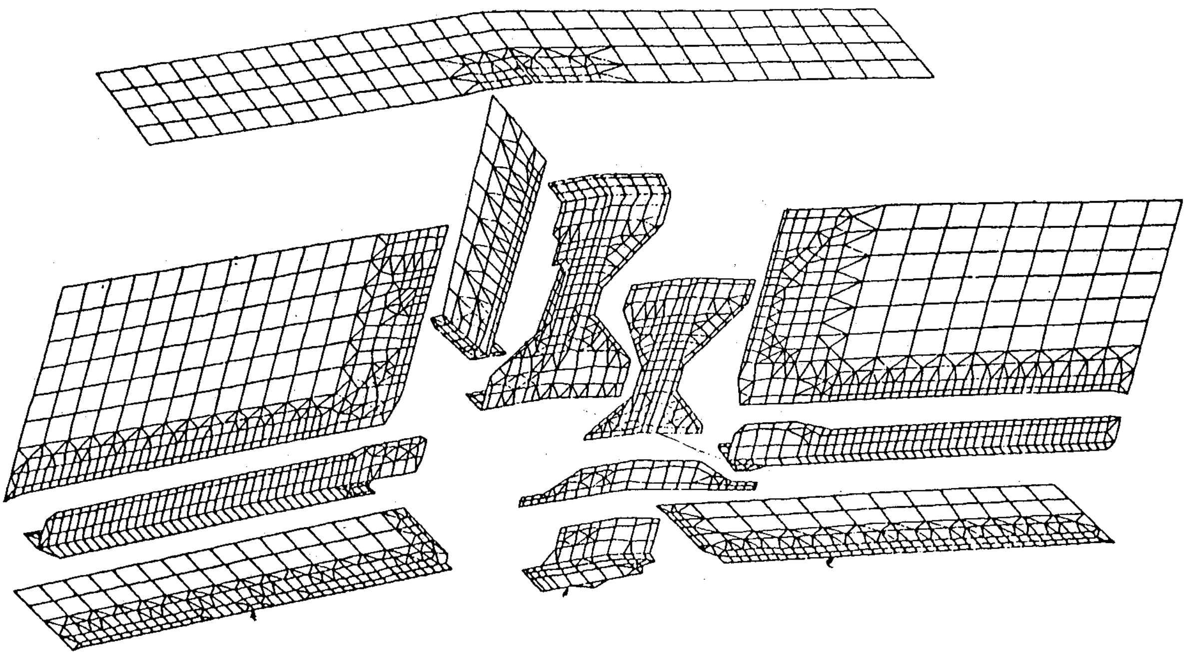

Fig. 7 C-141 WS 405 joint finite element model.

adhesive. To maintain the parallel with the Australian work, this same treatment is used in our repair procedure.

Structural Analysis

Preliminary sizing of repairs was made and the sizes were finalized using detailed finite element models (FEM) developed for each of the five candidate repair areas. These FEMs are analysed using the NASA Structural Analysis (NASTRAN) program. The FEMs are used to determine stress distributions in the structure with and without repairs. Wherever possible, existing FEMs are utilized, and only necessary modifications to include the effects of the composite repair are made. The anistropic properties of composite laminates are calculated by using a Lockheed-developed computer program. The bond in each FEM is modeled using linear elastic elements to determine bond shear stresses. The adhesvise bond is modeled as linear CFAST elements, and their stiffness is calculated separately. To more closely investigate the distribution and magnitude of these bond shear stresses, a detailed FEM was developed and different bond configurations studied. As example of these FEMS, those for WS 405 and FS 998 will be discussed.

The WS 405 repair FEM is a modification to the wing substructure of the C-141B general airframe FEM. Substantial detail is included in the rear beam, lower surface, and WS 405 area. Individual components in the model are upper and lower surface, spar webs, and caps, WS 405 fitting, failsafe strap, tiedown fitting, splices, and IWBRS 374 web. Fastener elements are included in this area to investigate load transfer distribution and their magnitude. Structural elements used in the detail model are three and four node membrane (CTRMEM and CQDMEM), semi-monocoque membranes (CSMTRM and CSMQDM), axial elements (CRODS), beam element (CBAR), and fastener elements (CFAST). The FEM results indicated that the benefit due to the addition of a boron/epoxy doubler gave a fatigue improvement ratio for the various elements of the joint conservatively calculated to be 2.4, including the effects of residual stresses due to differential thermal expansion between the doubler and the structure during curing. Figure 7 shows the FEM that was used.

At the FS 998 fuselage/main landing gear frame location, a FEM was developed to evalute the effect of a composite doubler spanning a crack in the web-to-cap radius of the FS 998 mainframe. Three model configurations were analyzed:

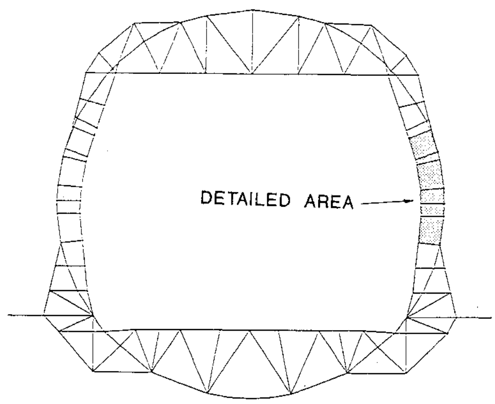

Fig. 8 C-141 FS 998 mainframe finite element model.

1) uncracked (baseline), 2) cracked, and 3) repaired. Axial and shear stress levels for the three configurations were then compared. The analysis is carried out in two steps. The model for the structural analysis of the FS 998 mainframe is run with design loads, and boundry loads from it are extracted and applied to a 3-dimensional model. Figure 8 shows the frame model and the location of the detailed model. The 3-dimensional detailed model of the FS 998 mainframe crack area consists of the structure from W.L. 183 to W.L. 213. Figure 9 shows the detail model, placement of composite doublers, and location of the 2.0-in. crack used in the cracked and repaired configurations. Two (1 forward, 1 aft) back-to-back composite angle doublers are attached to the frame web and inner cap. The doublers are modeled using membrane-type elements that match the geometry of the frame web and inner cap. Clearly, the main goal of composite repair of metal structures is to reduce stress levels and thereby slow crack initiation and growth. Consequently crack growth analyses were performed for each of the selected locations in order to estimate repairs benefit. As the WS 405 repair a good benefit of 2.4 was found, but an even better ratio was determined for the weep hole location as shown in Fig. 10. This figure graphically presents the effect on repair life from residual stress induced by temperature changes. For the wing weep hole repair area, the life at room temperature is reduced by 70% when subjected to a temperature differential of 135°F but is still a factor

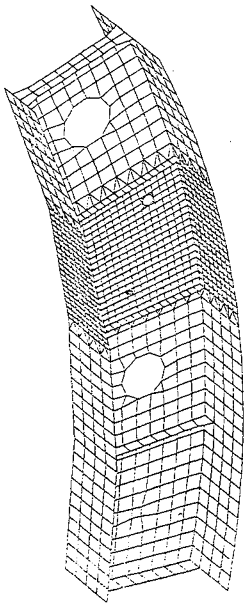

Fig. 9 C-141 FS 998 mainframe detailed finite element model.

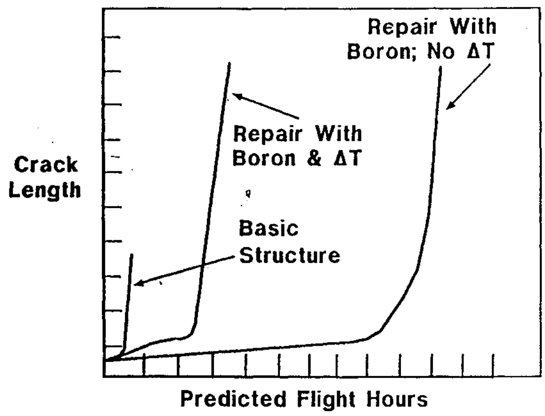

Fig. 10 C-141 weep hole analysis.

of 6 over the unrepaired structure. Of course this emphasizes the obvious advantage that can be derived from development of lower temperature curing adhesive systems.

Application Demonstration

Each of the selected locations had a demonstration conducted to prove the repair concept. The typical riser (weep hole), the lower surface panel (WS 77), and the main frame patch were all done on the original C-141 static test article (aircraft #6000) in storage at LASC-GA. Since the static test no longer has structure at WS 405, that location was simulated by a C-141 pylon location that had comparable compound curvature. The vertical stabilizer patch was installed on an actual in-service C-141B aircraft which was at LASC-GA for center wing repair. Another C-141B aircraft also at LASC-GA for center wing repair provided an opportunity for a new repair location.

A $1\frac{1}{2}$-in. fatigue crack was discovered which was repaired using boron/epoxy doublers. This repair required 3–4 weeks less aircraft down time and saved several hundred manhours. More importantly, however, this repair pioneers this type of technology for USAF aircraft which holds great promise for future force management benefits. To properly achieve the potential of this technology a test program to provide design and application verification is necessary to substantiate the analytical process already conducted. The test program envisioned will be at minimum cost, taking advantage of Australian knowledge and finite element modeling to correlate with simple tests. These tests, combined with in-service aircraft experience, will strengthen confidence in composite repair usage and lead to a full realization of this technology.

Conclusions and Recommendations

The objective to show the feasibility of concept use of composites materials for repair of aluminum structure was achieved. Four repairs were installed on the static test article, two repairs were installed on in-service aircraft, and one on a pylon. The material evaluation shows the most suitable applications to be graphic/epoxy and boron/epoxy composites. The material processes evaluation showed that the American Cyanamid FM73M structural film adhesive provides the needed durability and strength at 80°C (180°F) cure temperature. The simple surface treatment consisting of surface abrasion followed by A-187 silane treatment is effective for boron/epoxy and graphite/epoxy systems.

The concept of use of composites to repair metal structures has been determined to be feasible, and procedures have been developed for its installation. The following recommendations are made to extend the concept:

1) Bonding and sealing of materials should be evaluated for their long-term performance in the presence of aircraft fluids, moisture, temperature etc. Environmental tests should be conducted.

2) Newly developed bonding and sealing materials offering lower cure temperature and simpler surface preparation should be evaluated for their effectiveness in long-term performance.

3) Procedures should be developed so that the repair members can be prefabricated and be installed at field level.

4) The detailed FEM analysis of the repair predicted the stress distribution in structure with and without repair. These predicted stress reductions are used to project the benefit in reduced inspections and increased durability. This analyis should be verified by tests. This will show the need, if any, for refinements in methods of analysis.

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