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LARGE AREA COMPOSITE STRUCTURE REPAIR

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March 1979

Final Report For Period January 1976 - February 1979

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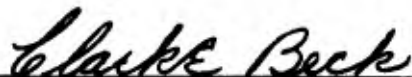
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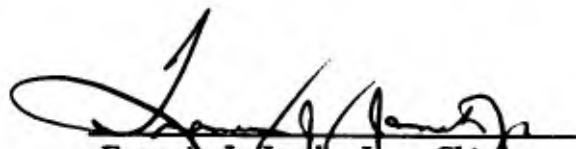
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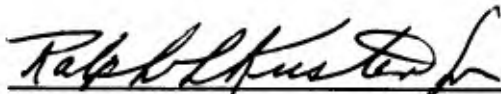


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20. ABSTRACT (Continue on reverse side if necessary and identify by block number) This final report summarizes activity for the period from 1 January 1976 to 22 February 1979 during a program aimed at developing and validating procedures for the repair of large-area damage to various graphite/epoxy and graphite-boron/epoxy primary and secondary structures. The effort was con- cerned with repair concepts that supply nearly complete structural res- toration to the damaged part, including the effects of temperature, moisture, (Continued)		

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and fatigue, while at the same time being amenable to military depot implementation.

Repairs were developed which were essentially flush on the outer surface to meet severe aerodynamic smoothness criteria, and which could be made to both monolithic and sandwich panels with access from either both sides or from one side only. Procedures were developed and evaluated using a low resin content (35%) AS/3501-6 unidirectional graphite/epoxy prepreg and FM 400 film adhesive as the primary repair materials with a no-bleed layup, cocured under vacuum pressure for two hours at 350F.

Testing was conducted to evaluate the effects of several adverse environments, including hydraulic fluid, jet fuel, paint stripper and water. Tensile tests indicated that water caused the greatest strength loss, and exposure to water was used during the program to evaluate repairs.

Twelve types of small scale specimens were evaluated, differing in material or repair configuration. Various test conditions were used including tension and compression, static and fatigue, -65F, room temperature and 265F, and both with and without moisture conditioning after the repair was made. Test results generally showed the repairs restored over 80 percent of the parent laminate ultimate allowable fiber failure strength.

Repair configurations were scaled up to intermediate size panels (12 x 24 inches), and then demonstrated by repairing five large scale panels which were subsequently tested as covers of box beams. All five panels were tested at room temperature with no moisture conditioning and sustained realistic design ultimate loading conditions with no failure. When loaded to failure in tension by beam bending, failures occurred at strains from 6900 to nearly 10,000 μ in/in, well above ultimate design values currently being used.

A final demonstration of the repair techniques was made by repairing a damaged F-15 graphite/epoxy speedbrake. The completed repair was proof loaded to 120 percent of limit load with no failure. Motion pictures were taken of each step in the repair process.

This program has demonstrated that advanced composite structures with large areas of damage can be repaired to restore between 80 and 100 percent of the parent laminate allowable fiber failure strength, that the repairs will be durable for the life of the aircraft and can be done at the Air Force depot maintenance facilities.

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PREFACE

This final report was prepared by the Northrop Corporation, Aircraft Group, Hawthorne, California covering work done under USAF Contract F33615-76-C-3017 between January 1976 and February 1979. The contract was administered by the Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio. Mr. Clark E. Beck ^(NH) was the Air Force Project Engineer. An accompanying document, AFFDL-TR-79-3039 was also prepared during the contract. It is a Repair Guide for Large Area Composite Structure Repair.

The program was conducted under the technical supervision of Dr. R.M. Verette, Manager of the Northrop Structural Mechanics Research Department. The following Northrop personnel made major contributions to the program.

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This final report was submitted for Air Force review on February 22, 1979.

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SUMMARY

An experimental program was conducted to develop and verify procedures for the repair of large damaged areas in advanced composite structural panels at depot level facilities. Procedures were based on criteria which required high strength restorations at an operating temperature range from -65°F to 265°F , with durability under fatigue loading and adverse environmental conditions. Repairs were developed which were essentially flush on the outer surface to meet severe aerodynamic smoothness requirements, and which could be made to both monolithic and sandwich panels with access from either both sides or from one side only.

Procedures were developed and evaluated using AS/3501-6 unidirectional graphite/epoxy prepreg and FM 400 film adhesive as the primary repair materials to repair parent laminates made of AS/3501-5 and T300/5208 laminates. A low resin content prepreg (35 percent) was used for the repair material with a no-bleed layup, cured under vacuum pressure for 2 hours at 350°F with no postcure. The reduced strength of the vacuum bag cured material was compensated for by selectively providing extra plies of repair material. Most repair configurations were made by cocuring the repair plies and the adhesive in place on a prepared scarfed surface with extra surface plies extending slightly outside the outer surface. A separate configuration was developed for a precured external patch, bonded over the damaged area after damage removal.

Testing was conducted to evaluate the effects of several adverse environments, including hydraulic fluid, jet fuel, paint stripper and water. Exposure was made both before and after bonding of a scarf joint made with FM 400 film adhesive. Tensile strength tests indicated that water caused the greatest strength loss, and exposure to water was used throughout the program to evaluate repairs.

Twelve types of small scale specimens were evaluated, differing in material or repair configuration. Various test conditions were used including both tension and compression, static and fatigue, -65°F , room temperature and 265°F , and both with and without moisture conditioning after the repair was made. Eight of the types involved nearly-flush repairs of monolithic panels using a modified scarf joint, three of which were made with access from one side only. Two of the types involved

configurations for precured external patches, and two were flush surface repairs on honeycomb sandwich panels with access from one side only. Test results generally showed the repairs restored over 80 percent of the parent laminate ultimate allowable fiber failure strength.

Repair configuration testing was scaled up to intermediate sized panels (12 x 24 inches) in which a 4-inch hole was repaired, to evaluate the effect of the local stiffness changes at the localized patch area. These tests demonstrated the advantage of flush repairs over external patches because of less change in localized stiffness and provided further insight into stacking sequence effects. Flush repairs restored over 95 percent of the parent laminate allowable fiber failure strength in both tension and compression for these intermediate sized panels. External patches restored 81 percent in tension and 66 percent in compression.

As a demonstration of the realistic application of the repair techniques, five large scale box beam covers were repaired and tested to failure. Each test panel was 60 inches long and 19 inches wide and was loaded as one cover of a 7-inch deep beam. The 60-inch long test section was the center bay of a 17-foot long beam loaded in four point bending and torsion. The test panels were attached to the aluminum substructure with mechanical fasteners. Two panels were 1/2-inch thick honeycomb sandwich, one of which was repaired with access from both sides, the other with access from one side. The other three covers were monolithic panels, two with part-through repairs and one with repair of a through hole. One of the part-through repaired monolithic panels contained inlaid strips of boron/epoxy, simulating the horizontal stabilizer of the B-1 aircraft. All five panels were tested at room temperature with no moisture conditioning, and sustained realistic design ultimate loading conditions with no failures. When loaded to failure in tension by beam bending, failures occurred at strains from 6900 to nearly 10,000 μ in/in, well above ultimate design values currently being used.

A final demonstration of the repair techniques which had been developed was made by repairing an F-15 graphite/epoxy speedbrake. Repairs were made using hand tools, vacuum bag cure and a heating blanket to approximate several conditions which would be typical of an on-aircraft repair. After replacement of a 40-square-inch section of aluminum honeycomb core and making flush repairs to 5- and 6-ply face sheets, the completed repair was proof loaded to 120 percent of limit load with no failure. Motion pictures were taken of each step in the repair process to illustrate the equipment and procedures used.

During the development and application of repair techniques, actual working times were recorded and compared with predicted values for both repairs and for initial fabrication. Data tend to show that for most parts, the cost of repairs would be much less than the cost of replacement parts. The tools, equipment and personnel skills required for the repair procedures as developed have been compared to those currently available at Air Force depot maintenance facilities (ALC's). Only very minor requirements for additional tools or equipment have been identified, although some training of both engineering and shop personnel is recommended.

This program has demonstrated that advanced composite structure with large areas of damage can be repaired to restore between 80 and 100 percent of the parent laminate allowable strength, that these repairs will be durable for the life of the aircraft and can be done at Air Force depot maintenance facilities. Specific procedures and repair joint configurations have been developed and demonstrated. Recommended procedures and configurations have been published in a Repair Guide for large area composite structure repair.

SECTION I
INTRODUCTION

The Large Area Composite Structure Repair program has been conducted to develop and verify procedures which can be used to repair large areas of composite structures. Previous to this program, composite structure repair methods had been limited to specific applications of limited size.

As early as 1969, Air Force-funded development work⁽¹⁾ addressed the problems of the repair of small area damage to lightly loaded boron/epoxy structures. Recently, repair techniques for small area damage have also been developed for graphite/epoxy and hybrid composite structures⁽²⁾, and field repairs for graphite/epoxy laminates have been developed using titanium alloy patches and mechanical attachments⁽³⁾.

This program has concentrated on the development of repair procedures for use at depot level facilities where relatively highly skilled personnel and adequate facilities and equipment are available. However, many of the techniques may be applicable at the organizational and intermediate (O & I) maintenance levels. All repairs are considered to be permanent and durable for the life of the aircraft. No repairs have been specifically developed for temporary use.

The term "large area " has been used to designate damage exceeding the previous state-of-the-art repair capability of approximately 2 to 3 inch diameter. The maximum size of repair actually made in the program was 12 inches in diameter, although the techniques developed are generally applicable to any size for which repair, rather than replacement, is selected. Most of the work has been done for graphite/epoxy panels, although repairs have also been developed for a graphite-boron/epoxy hybrid

1. G. Lubin, et al., Repair Technology for Boron/Epoxy Composites, Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio 45433, AFML-TR-71-270, February 1972. (Grumman Aerospace Corp.)
2. R. M. La Salle, Repair Procedures for Advanced Composite Structures, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-76-57, December, 1976. (General Dynamics Corp.)
3. James C. Watson, Field Repair of Composite Structures, Naval Air Development Center, Warminster, PA. (McDonnell Aircraft Co.)

panel configuration. Both monolithic and honeycomb sandwich configurations have been used. Repair of substructure was not a part of this program, although the effect of substructure and related systems components on accessibility were considered, and techniques were developed for both on-aircraft and off-aircraft repairs.

The repair procedures are intended to be generic in nature, i. e., applicable to a wide range of hardware, not limited to any single configuration. However, to ensure that the procedures would be applicable to emerging weapon systems, rigorous criteria for strength, temperature, aerodynamic smoothness, adverse environmental exposure and long term durability were imposed, as described more fully in Section II. To assist in making decisions of whether to repair or replace components, cost data were collected on specific large area repairs, and cost prediction methodology was employed.

The development of repair procedures is described in Section II, including the description of the criteria imposed, the generation of material and process data with environmental effects, and the development of repair scarf joint configurations. The resulting scarf joint was used extensively in several configurations for which small scale specimen tests are described in Section III, along with results for an alternate external patch configuration. The scale-up of repairs and their application to five large scale box beams and to an F-15 speedbrake which was damaged in service are described in Section IV. Overall conclusions and recommendations are given in Sections V and VI.

Much of the work done in the program has been documented previously in detailed interim reports issued as AFFDL Technical Reports ⁽⁴⁾ ⁽⁵⁾ ⁽⁶⁾. Where possible, information has been summarized in this final report with reference made to the more detailed information in the previous reports.

4. A. L. Scow, et al., Large Area Composite Structure Repair - First Interim Progress Report, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio 45433, AFFDL-TR-77-5, January 1977. (Northrop Corp.)
5. R. W. Kiger and S. H. Myhre, Large Area Composite Structure Repair-Second Interim Progress Report, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-77-121, November 1977.
6. R. W. Kiger and S. H. Myhre, Large Area Composite Structure Repair-Third Interim Progress Report, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-78-83, July 1978. (Northrop Corp.)

SECTION II

DEVELOPMENT OF REPAIR PROCEDURES

The repair procedures which were developed in this program were influenced by the relatively rigorous criteria for strength, temperature, aerodynamic smoothness, adverse environmental effects and long term durability, as described below. Use of these criteria ensured that the procedures developed would be applicable for components of aircraft to which these rigorous criteria apply. In cases where less severe criteria apply, these procedures can still be used or can be modified accordingly.

This section describes the criteria used for the program and their effect on the materials and processes selected and on the development of the single scarf joint used throughout the program. The work described in this section was a portion of Phase III of the program, and has been reported in more detail in the first and second interim progress reports (4, 5). The information presented here is intended to summarize the previously reported data and give the final results.

2.1 CRITERIA FOR REPAIRS

Restoration of Strength and Stiffness

The first requirement of any structural repair is the restoration of the necessary strength and stiffness of the part being repaired. This does not necessarily mean complete restoration of the parent material strength for fiber-dominated failure. Many airplane structures have dynamic stiffness, stability or fatigue requirements which limit the design ultimate stress or strain to a value significantly lower than the material failure strength. Static strength restoration must exceed the design ultimate strength and is intended to approach the parent material strength as nearly as possible within the limits imposed by other criteria. For this program, the B-basis (90% probability, 95% confidence) value for fiber failure of the parent material is employed as the reference value for evaluating strength performance. This is essentially the upper limit of probable strength criteria, and about 80% of this value would generally be considered adequate for a permanent repair.

Stiffness restoration should approximate the parent material stiffness since substantial stiffness deficiency or excess could be detrimental to the performance of the part.

Aerodynamic Smoothness

Design requirements for high performance aircraft typically include aerodynamic smoothness criteria which permit only minor local variations of the surface outside or inside the theoretical outer mold line (OML). Since some portions of the aircraft are more aerodynamically critical than others, varying criteria are usually applied to zones of the aircraft. For the most critical zones of the F-5A aircraft, for which detailed data were given in the first interim progress report⁽⁴⁾, a step in the surface is permitted up to 0.02 inch. Greater out-of-mold-line variations can occur if they are ramped with a slope not to exceed 4 in 100. Specific areas of higher performance aircraft may require even more restrictive criteria.

These criteria are established to govern the initial design of the overall configuration of the aircraft. A limited number of small isolated exceptions to the criteria in some locations would generally not affect the performance of the aircraft significantly, and therefore would be acceptable for individual repairs. For this program, repair configurations were developed for both external patches and for nearly flush repairs to satisfy a range of smoothness criteria.

Weight Growth

No specific criterion has been established on weight growth but it is obviously a matter of concern. In control surfaces, a repair would be evaluated against the critical weight and balance criteria for the specific part.

Environmental Compatibility

Two environmental conditions which are known to affect the strength of composite materials are temperature variation and moisture absorption. Evaluations of repairs have been made at -65°F, room temperature and 265°F with both dry and moisture-conditioned repairs. "Dry" means that the repair assembly was dry but was made with a parent laminate which had been conditioned in water at 180°F until it reached approximately a 1 percent weight increase. All parent laminates were moisture-conditioned before the repair was made. "Wet" means that the repair assembly was conditioned

by exposure to an environment of 95% relative humidity at 140°F for 30 days before test. The selection of materials and processes for this program was influenced by the 265°F temperature requirement and the need for adequate strength in the presence of moisture. The fact that both "dry" and "wet" repairs involved moisture conditioned parent laminates is a reflection of the fact that most, if not all, real repairs will be done on laminates which will have been in service for some time, and therefore will have absorbed some moisture prior to the repair.

Durability

Although consideration was originally given to temporary repairs, the procedures and configurations as developed are intended to be permanent and durable for the life of the aircraft. Spectrum load testing was used to verify the durability of the repairs.

Accessibility

In many actual repair situations, a choice will be available whether to make the repair on the aircraft or to remove the component for an off-aircraft repair. This decision is based on factors such as ease of removal, down time requirements, spares availability, etc. In order to provide repair procedures which can be used in either case, methods have been developed for making repairs with both one side access and two side access.

2.2 MATERIAL AND PROCESS SELECTION

This section summarizes the materials and processes selected for this program, the environmental compatibility studies which were made, and the design strength values used for design of repair configurations. The work discussed here was done as a part of Phase III of the program and has been reported in more detail in the first and second interim progress reports. (4, 5)

Parent Laminate Material

Because of its intended use on major Air Force hardware programs at the time this program originated (primarily the B-1 aircraft), AS/3501-5 unidirectional tape prepreg cured at 100 psi was specified for the parent laminates to be repaired in this program. A hybrid laminate having interleaved strips of boron using Avco 5505-4 boron/epoxy prepreg (also intended for use on the B-1) was used. These particular

material selections were made for the reasons cited; however, the program results are generally applicable to a variety of parent laminates.

Parent laminate strength values for a quasi-isotropic ($\pi/4$) laminate used extensively in the program were available from a previous Air Force program⁽⁷⁾. Limited testing was done in the current program using specimens conditioned to one percent moisture to compare with the previous data for specimens conditioned to approximately 1.8 percent moisture. For tension loading all test data fell within the usual scatter range, and so the previous results were accepted for the current program. However, the difference in the moisture contents appeared to affect the compression strength values significantly. Values for this program were therefore based on the limited testing done on one percent moisture specimens. Reference strength values were taken as 80 percent of the average values for compression. Since tensile values were based on a larger number of tests conducted in the previous program, actual statistical B-basis reference values were used. These reference values for the strength of 16-ply parent laminate material are listed in Table 1. The detailed test data from which these values were obtained have been given in the second interim progress report⁽⁵⁾.

TABLE 1. PARENT LAMINATE STRENGTH

Temp (°F)	Tension Allowable		Compression Allowable		Shear Allowable
	Loading (lbs/in.)	Strain (μ in./in.)	Loading (lbs/in.)	Strain (μ in./in.)	Loading (lbs/in.)
-65	6210	—	9740	—	—
RT	6860	11,000	7600	12,000	2836
265	5750	—	4710	—	—

Laminate stacking $[(\pm 45/0/90)_2]_s$ moisture content = 1.0%

Tension and shear allowables from reference 7

Compression allowables are 80% of average of three tests.

7. R. M. Verette and J. D. Labor, Structural Criteria for Advanced Composites, Final Report of Contract No. F33615-74-C-5182, Vol. 1, Air Force Flight Dynamics Laboratory, Wright Patterson Air Force Base, Ohio, 45433, AFFDL-TR-76-142, March, 1977. (Northrop Corp.)

Patch Ply Material

The material selected for the patch plies for this program was AS/3501-6 uni-directional tape prepreg. This material had potential strength advantages and, at the time of its selection, appeared to be potentially more available than the similar -5 material for long range future applications. In addition, it appeared to lend itself well to repair considerations as discussed below.

Repairs which are made in place on the aircraft will of necessity be limited to vacuum bag pressure cures, and visits to Air Force Depots early in the program indicated that autoclaves were not always available so that even off-aircraft repairs may be limited to vacuum bag curing. For these reasons, the decision was made that vacuum bag pressure cure cycles would be used for the repair procedures to be developed. Studies made early in the program indicated that the AS/3501-6 had higher strength for vacuum bag pressure than either T300/5208 or T300/934 materials.

For a repair patch, the number of plies normally will vary due to tapering of the patch near its edges. It was therefore desirable to use a no-bleed system to avoid the complication of trimming and varying bleeder plies. Prepreg with a resin content of 35 percent by weight was readily available and, after verification of properties using laminates cured with no bleeder plies, was used throughout the program.

Other material systems were not investigated for no-bleed curing during this program. Since resin systems vary in their sensitivity to cure cycle variations, the properties of other materials for patch plies should be investigated prior to their use, especially if the advantages of a low-resin, no-bleed system are desired.

The first step in developing an acceptable cure cycle for repair plies was to determine the effect of postcure times and temperatures on laminate properties. In order to isolate postcure effects from cure pressure effects, the laminates were cured at 100 psi. Postcure conditions of 2 hours at 400°F, 8 hours at 350°F, and no postcure were evaluated. Data are given in the first interim progress report⁽⁴⁾. There is no benefit due to postcuring on matrix sensitive properties such as transverse tensile, compression, and interlaminar shear strengths. This is true for specimens without moisture and for specimens containing 1 percent moisture by weight.

The second step in the cure cycle process development was to determine the effect of the heat-up rate. The rates evaluated were 1°F/minute, 3-5°F/minute, and 8°F/minute. Laminates were cured at vacuum bag pressure. Data from these laminates are also given in the first interim progress report⁽⁴⁾. The 8°F/minute heat-up

rate gave the poorest quality laminates and was rejected. Both the 1°F/minute and the 3°F/minute heat-up rates gave equivalent short beam shear strengths. The 3°F/minute heat-up rate gave superior laminate physical properties and was selected for the baseline cure cycle.

The following cure cycle was subsequently used for the patch material throughout the program:

1. Apply vacuum at room temperature.
2. Heat to 350°F at a 1-5°F/minute rate.
3. Hold at 350°F for 2 hours.
4. Cool to 150°F and release vacuum.
5. Do not postcure.

Design data were obtained for the AS/3501-6 patch material using the selected cure cycle and no bleeder plies with the low resin (35 ± 3%) prepreg. These data are given in detail in the first and second interim reports ^(4, 5), and are summarized here.

Physical properties were determined for the patch material. For cured laminates, resin content varied from 30 to 34 percent by weight, fiber volumes were between 55.3 and 58.7 percent, and void contents varied from 3.1 to 5.5 percent. These values are obviously less desirable than the 60 percent fiber volume and maximum 1 percent void content expected for the 100 psi cured parent laminate. Coupon tests indicated that the strength of the vacuum bag cured patch material tended to be slightly less than for the parent material. Development of repair configurations made allowance for the potential material strength reduction, as discussed in Paragraph 2.3.

A wet layup using a two-part resin system and unprepregged woven graphite fabric was also tried early in the program. A bisphenol F resin (Dow Chemical Company Epoxy Resin 7818) was used with Epon Z hardener. The detailed description of the fabrication and testing work was given in the first interim progress report⁽⁴⁾. The work was discontinued when results were not repeatable and unanswered questions occurred with suppliers concerning the correct cure cycle parameters to be used.

Adhesive

FM-400 was selected as the adhesive for this program. It is a widely used adhesive and has acceptable properties at temperatures up to 350°F. Considerable experience and technical data were available for this adhesive.

A thorough survey was made of several adhesive materials for this program. Obviously, it is highly desirable to use an adhesive which is currently being widely used in aircraft repair operations. Unfortunately these currently used adhesives are generally of the lower temperature type (maximum service temperature 180-200°F) which cannot meet the 265°F service temperature requirement of this program. Table 2 presents a matrix summarizing the advantages and disadvantages of the various adhesives.

TABLE 2. COMPARISON OF ADHESIVES

ADHESIVE MATERIAL	ADVANTAGES	DISADVANTAGES
I. 180-200°F service temperature, adhesives, e.g., AF-55, FM-73, AF-127	1. High strength, tough adhesives 2. Widely used	1. Can't meet 265°F service temperature requirement
II. 350-400°F service temperature adhesives, e.g., FM-400, AF-143, MB-329, RB-398, PL-731	1. Meets service temperature requirement 2. Widely used	1. Lower strength, less tough than 180-200°F adhesives

FM-400 was selected from the 350-400°F service temperature adhesives because of previous satisfactory experience with it. Consideration was also given to AF-147. This adhesive which has a maximum service temperature of 300°F, is a tougher, higher shear strength material than the 350°F adhesives and is attractive for that reason. However, it was not used in this program because of its limited availability and uncertain production commitment at the time the selection was made.

The shear strength of an adhesive is highly sensitive to the specimen configuration used to obtain strength values, e.g., torsion ring, rail shear, single or double overlap, etc., as well as to the test environment and cure cycle used. For preliminary configuration studies of bonded joints, an average value of 6640 psi for the shear strength was used, based on rail shear tests of the cured adhesive described in the first interim progress report⁽⁴⁾.

Compatibility Test Program

Composite aircraft structures are exposed to several potentially adverse environments during their service lives. To investigate the effect of these environments on repairs, specimens were exposed to four liquids: (1) water (moisture), (2) JP-4 jet

fuel, (3) hydraulic fluid (MIL-H-5606), and (4) paint stripper (Turco 5351). After a prescribed period of exposure, strengths of the exposed specimens were determined.

The details of the methods of exposure, bonding operations and strength testing are fully reported in the first interim progress report (4), and will be summarized here.

Tensile tests were made on the following 3 types of specimens:

1. Unidirectional parent material specimens and patch material specimens (containing no bondline in the test area) to establish baseline control values for the program. These tensile tests were performed on both exposed and unexposed control coupons.
2. Bonded parent-to-patch assemblies for which the unexposed patch material was bonded to parent material which had been exposed to the four environments previously listed, plus a control group with no exposure. These tests were run to determine quantitatively how well the patch material could be bonded to parent material which had been exposed to the four liquids.
3. Bonded parent-to-patch assemblies re-exposed after the bonding operation. Tests were run to establish how well the bond would survive exposure conditions in field service to moisture, JP-4 jet fuel and hydraulic fluid. The bonded assemblies were not exposed to paint stripper since aircraft structures are normally exposed to paint stripper only during repair operations, not during normal service use.

The environmental exposures selected were a compromise between real life exposures and accelerated, very severe exposures. Moisture exposure was by immersion in 180°F water until the specimens absorbed approximately 1 percent by weight. This exposure was selected because surveys made worldwide show that composite structures achieve a typical moisture content of approximately 1 percent⁽⁸⁾. Exposures in JP-4 jet fuel and hydraulic fluid were 30 days at room temperature. Exposure in paint stripper was 24 hours at room temperature. Re-exposure after bonding was for the same times and conditions. The types of tensile coupons which were used in the program are described in Table 3 and shown in Figure 1.

8. Private communication with R. L. Stoecklin, Boeing Commercial Airplane Co. regarding unpublished data on 2 year exposure data showing approximately 1 percent maximum moisture absorption for worldwide environment.

At room temperature and at 265° F, the tensile strength of AS/3501-5 and AS/3501-6 unidirectional laminates was unaffected by exposure to moisture, paint stripper, JP-4 fuel and hydraulic fluid. The exposure to moisture produced specimens which broke on testing in the gage area or at the end of the tab. The end-of-the-tab failures appeared no different in mode of failure than the other specimens in the group. The exposure to environmental fluids in this compatibility program is believed to be

TABLE 3. COMPATIBILITY PROGRAM TEST SPECIMENS

Test	Laminate Orientation	Thickness or Plies	Joint Bond	Specimen Size	Composite Specimen Tabbed
Parent Tensile	0°	8	None	0.5" x 8"	Yes
Patch Tensile	0°	8	None	0.5" x 8"	Yes
Composite/Composite	0°	16/16	Scarf	1.0" x 8"	No
Composite/Titanium	0°	16/Ti-0.090"	Double Lap	1.0" x 8"	No

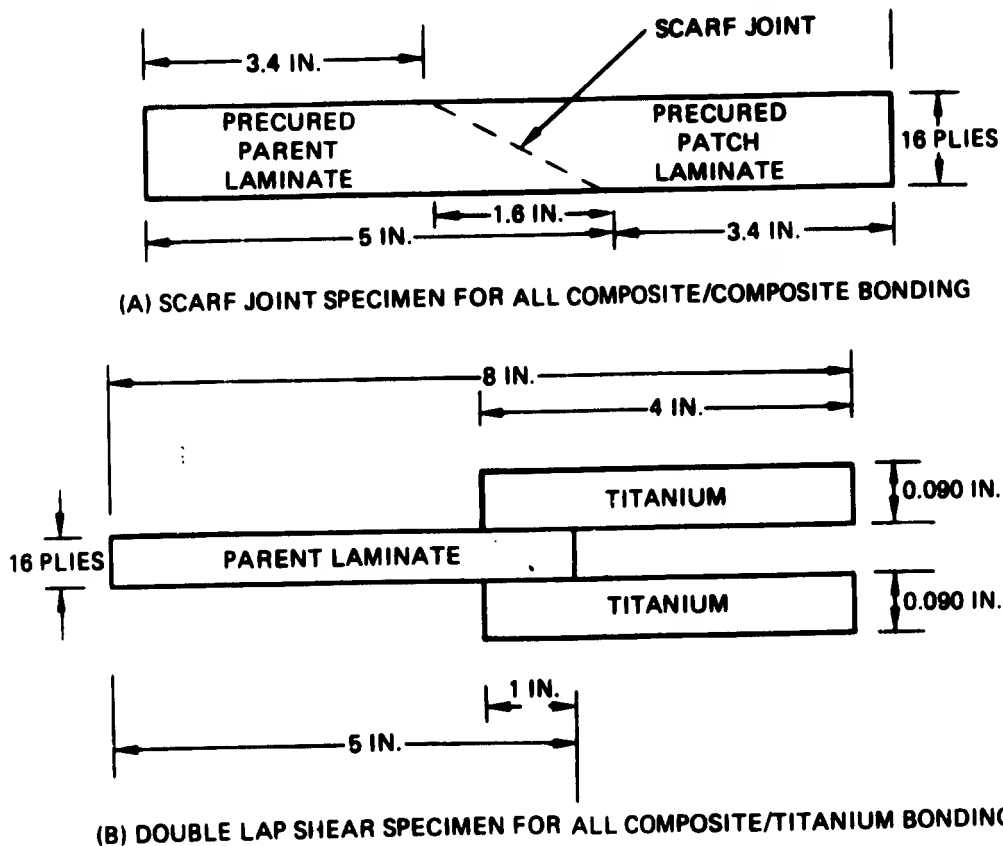


Figure 1. Configuration of Compatibility Test Coupons

much more severe than what might be expected in service. Since no distinct degradation of the parent or patch material was observed in these tests, none is expected under service conditions.

Moisture had a significant effect on the strength of precured scarf joints made with FM-400 adhesive and graphite/epoxy adherends. In the specimens in which only the parent material was exposed to moisture, the room temperature strength was unaffected but the 265°F strength was reduced by about 17 percent. In the re-exposed specimens, where the bonded specimen was exposed to environmental fluids, the reduction in strength due to moisture was 38 percent. This is believed to be an effect mainly on the adhesive. There was no degradation due to paint stripper. There was degradation in some tests due to JP-4 and hydraulic fluid.

Complete data are tabulated in the first interim progress report (4). Results are summarized graphically for precured scarf joints in Figure 2. Because moisture was the most severe environment, many of the specimens tested subsequently in the program were exposed to moisture prior to evaluating the strengths of the repairs.

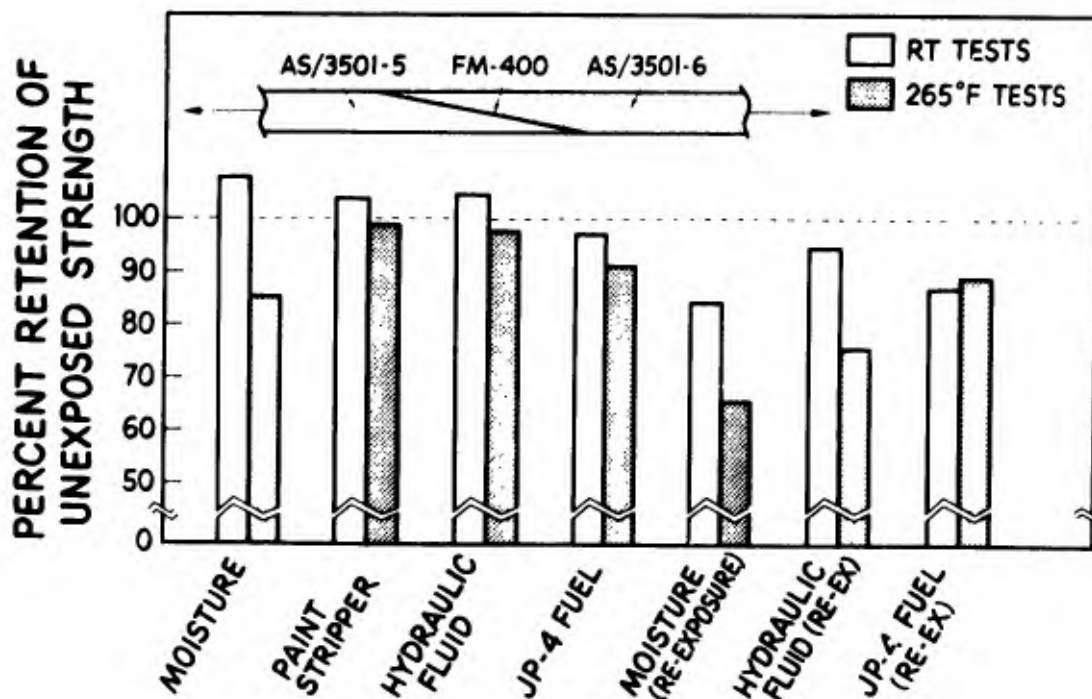


Figure 2. Scarf Joint Compatibility Test Results

2.3 SCARF JOINT CONFIGURATION DEVELOPMENT

The development of the bonded scarf joint which was used in nearly all repairs implemented during the program proceeded forward on three legs: design, analysis and test. The basic concept of the scarf joint is obvious and simple. As the thickness of one side of the joint (say the parent laminate) is diminished, the thickness of the opposite side (say 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 procedure is illustrated in Figure 3 for a flush scarf joint having identical 16 ply $[(0/+45/90)_2]_S$ laminates. The stiffness distribution in both laminates is given on the small bar chart to the left of the sketch. The values are functions of the laminate ply distribution and the lamina elastic properties. They can be found from the lamina stress SIGX printout of the SQ5 program of Reference 9. By assigning the thickness per ply the value of unity, the relative Et values of each adherend are determined by accumulating the relative E values at each station as thickness is increased by one ply per station increment. Knowing the Et of each adherend, the total Et is found and the relative load in one adherend determined from the assumption that the load in each adherend is proportional to the adherend Et divided by the total Et.

The difference in the load fraction at two adjacent stations is the average relative shear (R. S.) loading for the increment, i. e.,

$$(R. S.)_{i, i-1} = (L/T)_i - (L/T)_{i-1}$$

where L_i = Left Et of station i

T_i = Total Et of station i.

The relative shear diagram gives some insight into the load transfer mechanics. The higher shears exist on those steps where the 0-degree ply of the thinner adherend is bonded.

9. D. L. Reed, Point Stress Laminate Analysis, Report FZM-5494, General Dynamics, Fort Worth, April 1970.

The strength of the splice is inversely proportional to the maximum relative shear, i. e.,

$$P_{ULT} = F_{SU} (\Delta x) / M. R. S$$

where F_{SU} = adhesive ultimate shear strength, psi

Δx = station increment, inch

M.R.S. = Maximum Relative Shear

Adhesive shear strengths are strongly influenced by the kind of test performed. It has been found that shear strength determined by the rail shear test on FM-400 adhesive can be used in conjunction with the preceding analysis to adequately predict the strength of the splice.

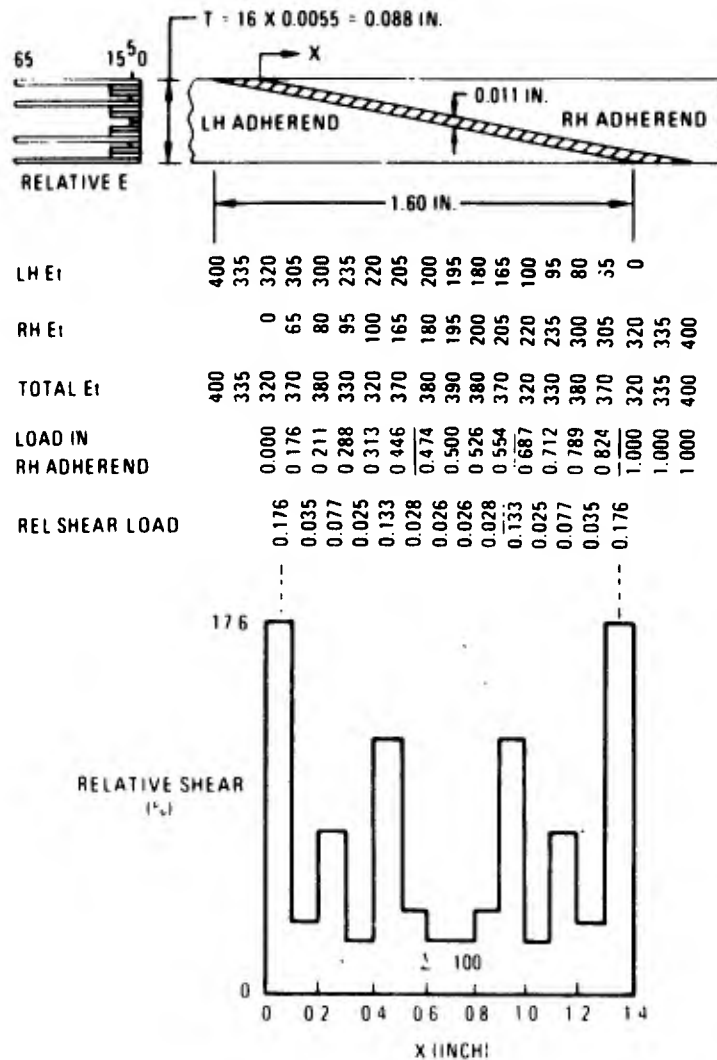


Figure 3. Shear Distribution Analysis for Scarf Joint with Identical $[(0/+45/90)_2]_s$ Adherends

A real repair joint is not like the idealization of Figure 3 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. Although the patch plies are not scarfed, they are assumed to transfer their load uniformly through the end 0.10 inch. Hence, for analysis the patch has been treated the same as the parent.

The scarf joint in Figure 3 is exactly flush on both surfaces and hence it is 0.20 of an inch shorter than the length of the scarf. The effect of increasing the overlap of the two adherends is to reduce the maximum relative shear to a value of 14 percent at 1.70 inches of lap. At this lap length the longest patch ply laps the shortest ply of the parent by 0.10 inch. No further reduction in maximum relative shear is possible by this method. This increased overlap joint became the basic cocured 16-ply scarf joint as shown in Figure 4. It consistently failed in the splice at about 3490 pounds per inch or 51 percent of the room temperature dry tension allowable of Table 1. The scarf length was selected at 0.10 inch per ply which is much shorter than has been commonly recommended. A longer scarf would reduce the average shear stress on any step of the splice, since the step would be longer. However, the actual peak stress at the end of the longest 0-degree patch ply (M. R. S. = 14 percent) would be little reduced since the 0.10-inch step length is 18 times the nominal ply thickness, which theory and experiment indicate to be quite sufficient to load the ply.

Special ply termination tests were conducted on the 16-ply $\pi/4$ laminate $[(0/\pm 45/90)_2]_s$ with an additional 0-degree ply laid up on each surface on each end of the coupon but not extending across the middle of the coupon. Each specimen therefore had four ply terminations extending across the specimen width, each at a different station. Using the "photostress" technique, the length required to materially change the strain due to one ply terminating was seen to be very small, i. e., on the order of a few hundredths of an inch. When tension load was applied, the control specimens with terminating plies cut off straight and perpendicular to the longitudinal axis failed at 77 percent of the parent laminate tension allowable. In each of three specimens, failure occurred by one or more of the terminating plies peeling away from the laminate. On the other hand, specimens with several kinds of serrations cut into the end of the terminating plies, as shown in Table 4, all failed near 100 percent of the parent tension allowable and failed in laminate tension. No peeling occurred in any of the 12 specimens

PATCH: AS/3501-6 COCURED

$[(0/45/90)_2]_S$

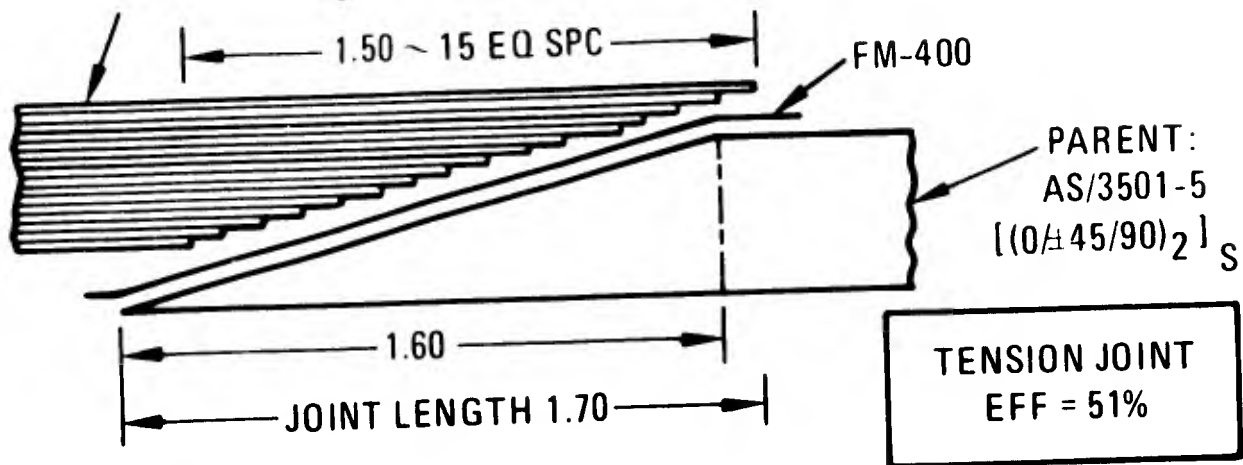

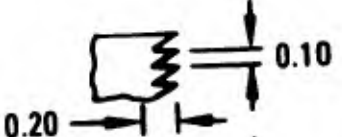




Figure 4. Basic Cocured 16-ply Scarf Joint Design

TABLE 4. RESULTS OF PLY TERMINATION TESTS

PLY END CUT	TENSION ULTIMATE STRENGTH (LB/IN)	MODE OF FAILURE	% OF PARENT ALLOWABLE
CONTROL 	5260	PEEL	77
	{ 6210 6633	TENSION TENSION	91 97
	7063	TENSION	103
	6900	TENSION	100

tested having a total of 48 serrated ply ends. (*) With a pair of commercial pinking shears a satisfactory serration can be cut easily. The implications of the ply termination tests are obvious: patch peeling can be prevented even with brittle adhesives, and a joint capable of the parent tension allowable may be possible.

The longest ply of the parent adherend may actually distribute its load to the adhesive much better than the longest ply of the patch since the parent ply is scarfed. Although this may be the design intent, the longest ply may in practice not be scarfed at all. If the scarf actually intersects the inner surface as it should, the edge will look a bit ragged. Shop people are inclined to trim off the ragged edge to give the work a neater look. Specific instruction must be given to scarf the laminate all the way to the inner surface and accept the feathered edge appearance. Another method of improving the load transfer at the end of the scarf is to shear the load through two surfaces rather than one.

Two small changes were made to the basic cocured scarf joint to verify the improvements expected from attention to the details at the splice ends. As shown in Figure 5, the longest patch ply has been extended slightly and a small patch of one 0-degree ply was added to the inside. Both plies had serrated ends. These changes were sufficient to change the mode of failure to patch laminate tension and to raise the joint efficiency to 70 percent. Recognizing the lower strength of the patch material and the eccentricity caused by the small inside patch, the results were very encouraging.

The final single scarf joint configuration, as shown in Figure 6, involved one additional modification. The generic parent laminate was assumed to be loaded in shear as well as longitudinal tension or compression. Therefore, one ply was added in each of three directions (0, +45 and -45) on both sides of the laminate. This joint developed 97 percent joint efficiency (room temperature, dry, tension) consistently failing the parent laminate. In retrospect, the short ply on the inside is probably not necessary.

(*) Correction to Reference 4, First Interim Progress Report, AFFDL-TR-77-5, Northrop Corp., Jan. 1977, Figure 19 erratum: the last entry failure mode should be (5) indicating laminate tension in 16-ply section.

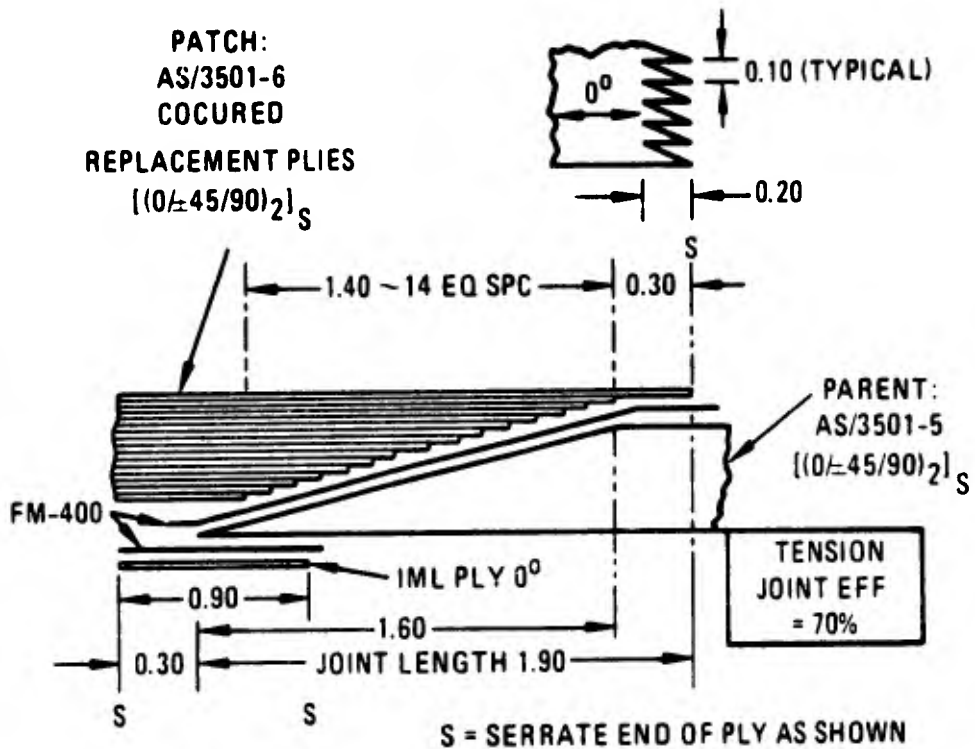


Figure 5. Refined Scarf Joint Design, Including Serrations

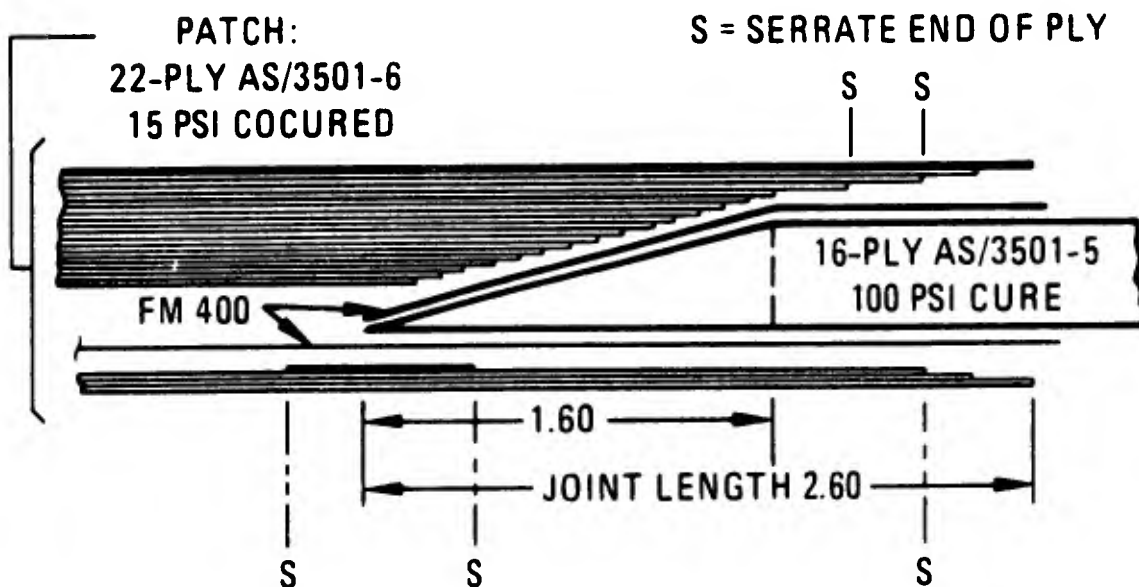


Figure 6. Final Single Scarf Flush Repair

SECTION III

REPAIR JOINT CONFIGURATION TESTING

The analytical and empirical development of the single scarf joint which was used for flush repairs was described in Paragraph 2.3. This section describes an extensive test program in which several variations of the single scarf configuration were investigated for one side access, or "blind" repairs, partial thickness repairs of thick laminates, alternate parent material, and repairs to sandwich panels. An external patch configuration was also investigated. Many of these were tested in both tension and compression, "wet" and "dry", and at elevated and reduced temperatures. Table 5 shows the configuration variations and the number of specimens tested for each test parameter. The test program was conducted as part of Phase III of the program, and most results were described in detail by specimen types 1 through 12 in the second and third interim progress reports. (5, 6)

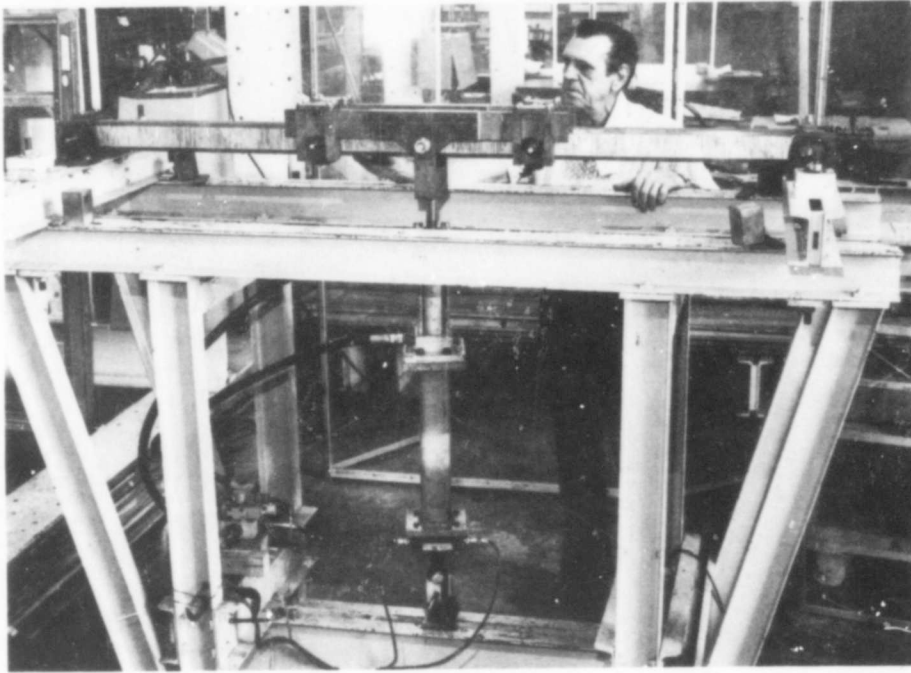
The test program is summarized in the following paragraphs. Flush repairs on monolithic panels (Types 1 through 7 and Type 12) are described in Paragraph 3.1, repairs to sandwich panels (Types 8 and 9) are described in Paragraph 3.2 and external patches (Types 10 and 11) are described in Paragraph 3.3. A separate series of tests using intermediate sized panels is described in Paragraph 3.4.

Some of the tension tests for flush monolithic panel repairs were conducted using tension coupons one inch wide, with the scarf joint for the repair running across the full width. The balance of the tests, including some tension and all compression tests, were done using sandwich beam specimens with four point loading so that the bending moment, and hence the load in the repaired face sheet, was constant over the center span. The sandwich beam test arrangement is shown in Figure 7.

Specimens were fabricated by first making the bonded repair to the face sheet, then moisture conditioning those so designated, and then bonding the face sheet to the core. The 16 ply repairs were tested as sandwich beams 1.0 inch wide and 26.0 inches long with 1.5 inch core. The 50 ply repairs were tested as sandwich beams 1.5 inches wide and 50 inches long with 2.0 inch core. The type 4 boron-graphite hybrid specimens

TABLE 5. REPAIR JOINT COUPON AND BEAM TEST MATRIX

SPECIMEN TYPE	MATERIALS	DESCRIPTION	PREPARE	DRY												MOISTURE CONDITIONED											
				STATIC TENSION			FATIGUE			STATIC COMPRESSION			FATIGUE			STATIC TENSION			FATIGUE			STATIC COMPRESSION					
				ST	CT	MT	ST	CT	MT	ST	CT	MT	ST	CT	MT	ST	CT	MT	ST	CT	MT	ST	CT	MT			
① S-PLY	AS 3001 S AS 3001 S FM 400	PRE-CURED CO-CURED	1	3	5	6	3	3	3	6	3	3	3	3	3	3	3	3	3	3	3	3	3				
② S-PLY	AS 3001 S AS 3001 S FM 400	PRE-CURED CO-CURED	—	6	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
③ S-PLY	AS 3001 S AS 3001 S FM 400	PRE-CURED & ROUTED CO-CURED	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6	6				
④ S-PLY	AS 3001 S AS 3001 S FM 400	PRE-CURED & ROUTED CO-CURED	3	3	3	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑤ S-PLY CO-CURED	AS 3001 S AS 3001 S FM 400	PRE-CURED CO-CURED	3	3	3	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑥ S-PLY SPLIT	AS 3001 S AS 3001 S FM 400	PRE-CURED CO-CURED	—	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑦ S-PLY VACUUM	AS 3001 S AS 3001 S FM 400	PRE-CURED CO-CURED	—	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑧ S-PLY	AS 3001 S AS 3001 S FM 400 H-C CORE H-C CORE	PRE-CURED CO-CURED	1	3	3	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑨ S-PLY	AS 3001 S AS 3001 S FM 400 H-C CORE	PRE-CURED CO-CURED	—	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑩ S-PLY	AS 3001 S AS 3001 S FM 400 H-C CORE H-C CORE	PRE-CURED CO-CURED	—	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑪ S-PLY	AS 3001 S AS 3001 S FM 400 H-C CORE H-C CORE	PRE-CURED CO-CURED	—	3	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—	—				
⑫ S-PLY	T3000/3200 AS 3001 S FM 400	PRE-CURED CO-CURED	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3	3				



78-03868-3

Figure 7. Sandwich Beam Test Arrangement

were 2.0 inches wide and 70 inches long with 2.0 inch core. In each case, the repair joint configuration ran across the full width of the specimen at one location at mid-span. Typical 16 ply specimens are shown in Figure 8.

The primary advantages of the sandwich beam configuration are the gradual load introduction into the compression face which results from the four point loading, and the stability provided to the compression face by the core. For convenience in fabrication, some of the tension specimens were also tested as sandwich beams by inverting the beam in the loading fixture.

For all tests described in this section, with the exception of Type 4 as discussed later, the parent laminate was moisture conditioned to approximately 1 percent moisture content prior to making the bonded repairs. For those specimens referred to as "dry", no further conditioning was done. Specimens designated to be "wet" were re-exposed to 95 percent relative humidity at 140°F for 30 days prior to testing.

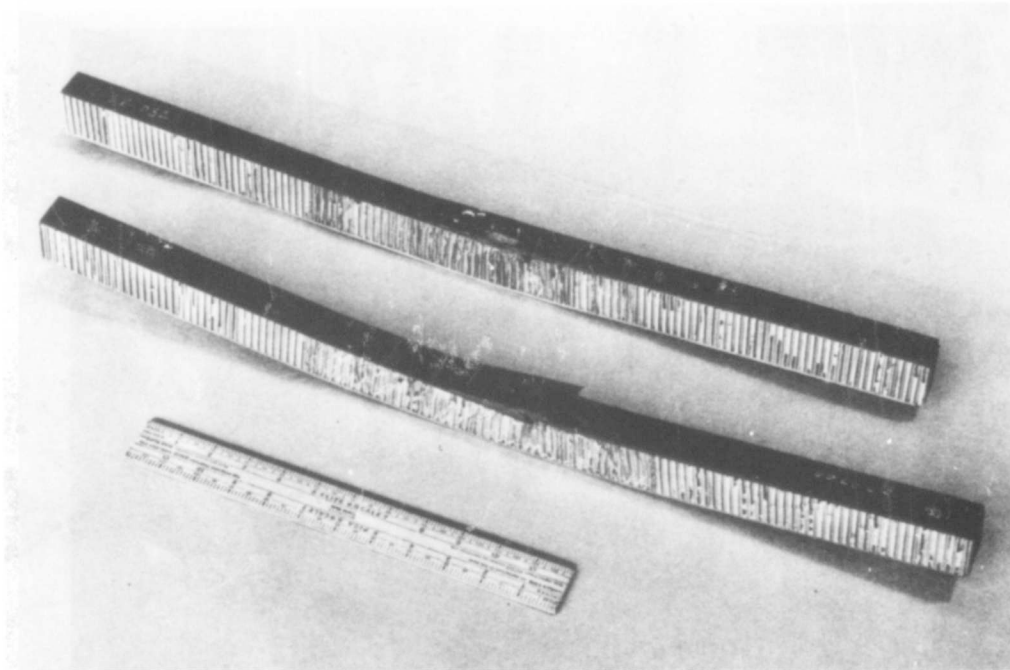


Figure 8. Sandwich Beam Specimens

78-03871-7

Those specimens designated for elevated (265°F) or reduced (-65°F) temperature testing were enclosed in a chamber into which heated or cooled air was blown.

Some specimens were exposed to fatigue loading prior to strength testing, using the same test arrangement previously shown in Figure 7. The fatigue spectrum used was developed for the lower wing skin near the root rib of the F-5E aircraft. The exceedance curve, shown in Figure 9, represents 1000 hours of flight time with compressive loads deleted. The flight-by-flight spectrum is randomized as to the ordering of the loads and sequencing of the flights. Two lifetimes, or 8000 flight hours, of loading were applied at room temperature to each fatigue specimen.

The maximum load in the spectrum was scaled to produce a peak strain of 4570 $\mu\text{in/in}$ in the specimens. This value of strain is the B-basis matrix failure strain for the parent laminate, and was considered to be an upper limit on actual spectrum loading based on current design practice. A lower value of peak strain, 2510 $\mu\text{in/in}$, was applied to a separate group of most specimen types to help identify effects of spectrum loading. Regardless of the direction of loading for residual strength tests, all fatigue loading was applied to cause tension in the specimens.

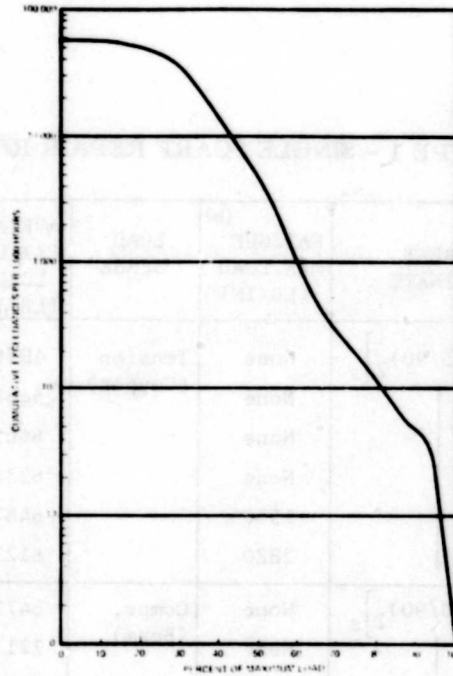


Figure 9. F-5E Load Spectrum For Wing Lower Skin at BP 29,5




3.1 FLUSH REPAIRS ON MONOLITHIC PANELS

The single scarf joint whose development was described in Paragraph 2.3 was used with variations for Types 1 through 7 and Type 12, as listed in Table 5 and described in the following paragraphs.

Type 1 - Single Scarf Joint

The development of the single scarf repair joint configuration was described in Paragraph 2.3 with the resulting design shown in Figure 10, Splice Detail G1. Fourteen of these specimens were tested as one inch wide tension coupons for the dry tension tests, series I, II, III and IV in Table 5. The results are shown in Table 6. The predominate failure mode was a tension failure in the replacement plies. Recognizing the lower strength of the vacuum-pressure-cured replacement plies, the six added plies, three on each face, were extended to the left end of the specimen, or in effect, across the entire repair as shown in Figure 10, Splice Detail G2. This raised the load at failure and changed the failure mode to tension failure in the parent laminate.

TABLE 6. TYPE 1 - SINGLE SCARF REPAIR RESULTS

TEST SERIES	(a) SPLICE DETAIL	TEST TEMP (°F)	PARENT LAMINATE	(b) FATIGUE MAX LOAD (LB/IN)	LOAD SENSE	(c) AVERAGE FAILURE LOAD (LB/IN)	(d) PARENT ALLOWABLE (LB/IN)	JOINT EFF
I	G1	-65	$[(0/\pm 45/90)_2]_s$ 	None	Tension (Coupon)	4970	6210	0.80
II	G1	RT		None		5688	6860	0.83
II	G2	RT		None		6665	6860	0.97
III	G1	265		None		6333	5750	1.10
IV	G1	RT		1550		6487	6860	0.95
IV	G1	RT	2820	6123	6860	0.87		
V	H	-65	$[(+45/0/90)_2]_s$ 	None	Compr. (Beam)	8471	9740	0.87
VI	H	RT		None		7213	7600	0.95
VII	H	265		None		6230	4710	1.32
VIII	H	RT		1550		7636	7600	1.00
VIII	H	RT		2820		7830	7600	1.03
IX	H	-65	$[(\pm 45/0/90)_2]_s$ 	None	Tension (Coupon)	6706	6210	1.08
X	H	RT		None		6890	6860	1.00
XI	H	265		None		5057	5750	0.88
XII	H	RT		1550		6575	6860	1.00
XII	H	RT		2820		7136	6860	1.04
XIII	H	-65		None	Compr. (Beam)	8076	9740	0.83
XIV	H	RT		None		5988	7600	0.79
XV	H	265		None		4946	4710	1.05
XVI	H	RT		1550		6830	7600	0.90
XVI	H	RT		2820		7165	7500	0.94

(a) Splice details shown in Figure 6.

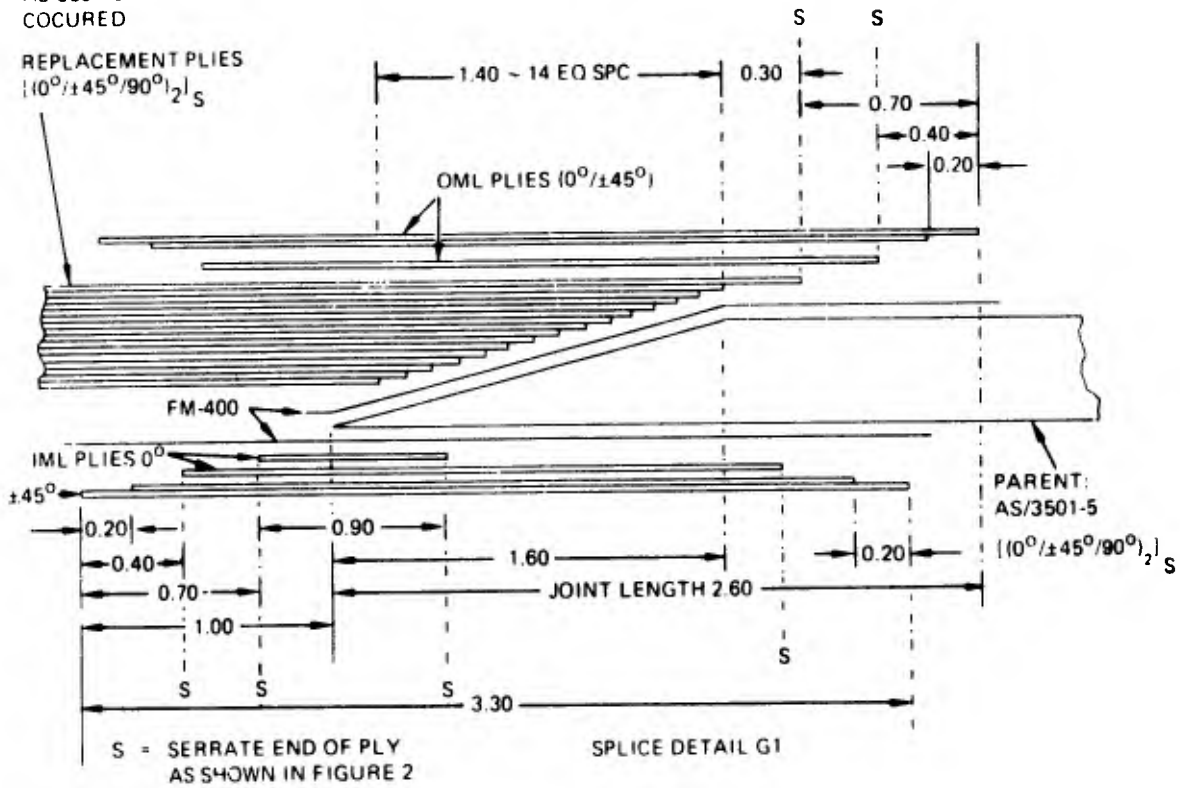
(b) Fatigue Loading for F-5E Wing, two lifetimes.

(c) Average of three replicates.

(d) Parent Laminate Allowables from Table 1.

PATCH
AS/3501-6
COCURED

REPLACEMENT PLYS
 $[(0^\circ/\pm 45^\circ/90^\circ)_2]_S$



PATCH:
AS/3501-6
COCURED

REPLACEMENT PLYS
 $[(0^\circ/\pm 45^\circ/90^\circ)_2]_S$

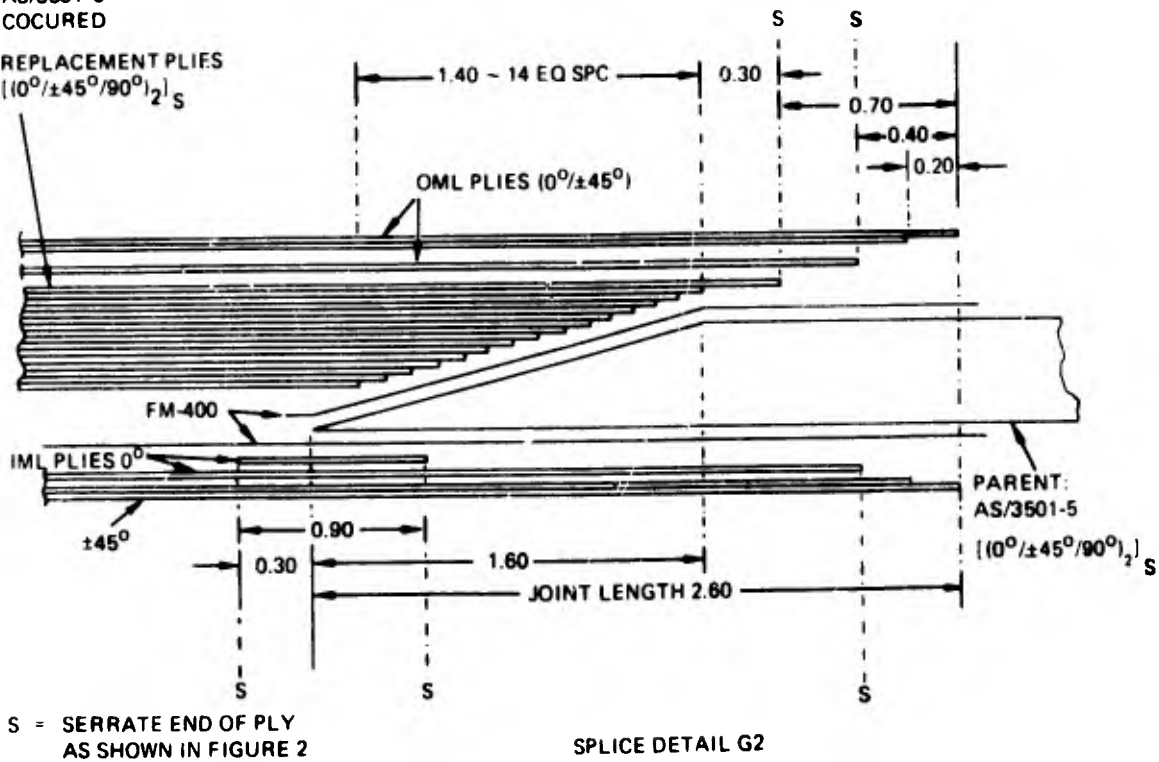


Figure 10. Single Scarf Specimen Type (1) Detail G

Compared to a parent allowable of 6860 lb/in, the joint efficiency is 97 percent. This is considered to be a very satisfactory repair design.

For the remainder of the Type 1 tests, the parent laminate stacking order was changed to put ± 45 plies on the surfaces, which is more representative of current design practice. The stacking sequence of the replacement plies was changed accordingly, which necessitated other minor changes, resulting in the splice detail shown in Figure 11, Splice Detail H. The hand-cut serration of the ends of some of the 0-degree plies was changed to a 1/8-inch-deep cut with commercial pinking shears. Specimens were then produced in sufficient quantity for the remaining tests of Specimen Type 1. All tension specimens were 1.00-inch-wide coupons, and compression specimens were the compression faces of 1.00-inch-wide sandwich beams.

Test results are tabulated in Table 6 and compared graphically in Figure 12. Note that room temperature tension tests (Series II) were done on both Splice Details G1 and G2. The increased strength of Detail G2 resulted in a joint efficiency increase from

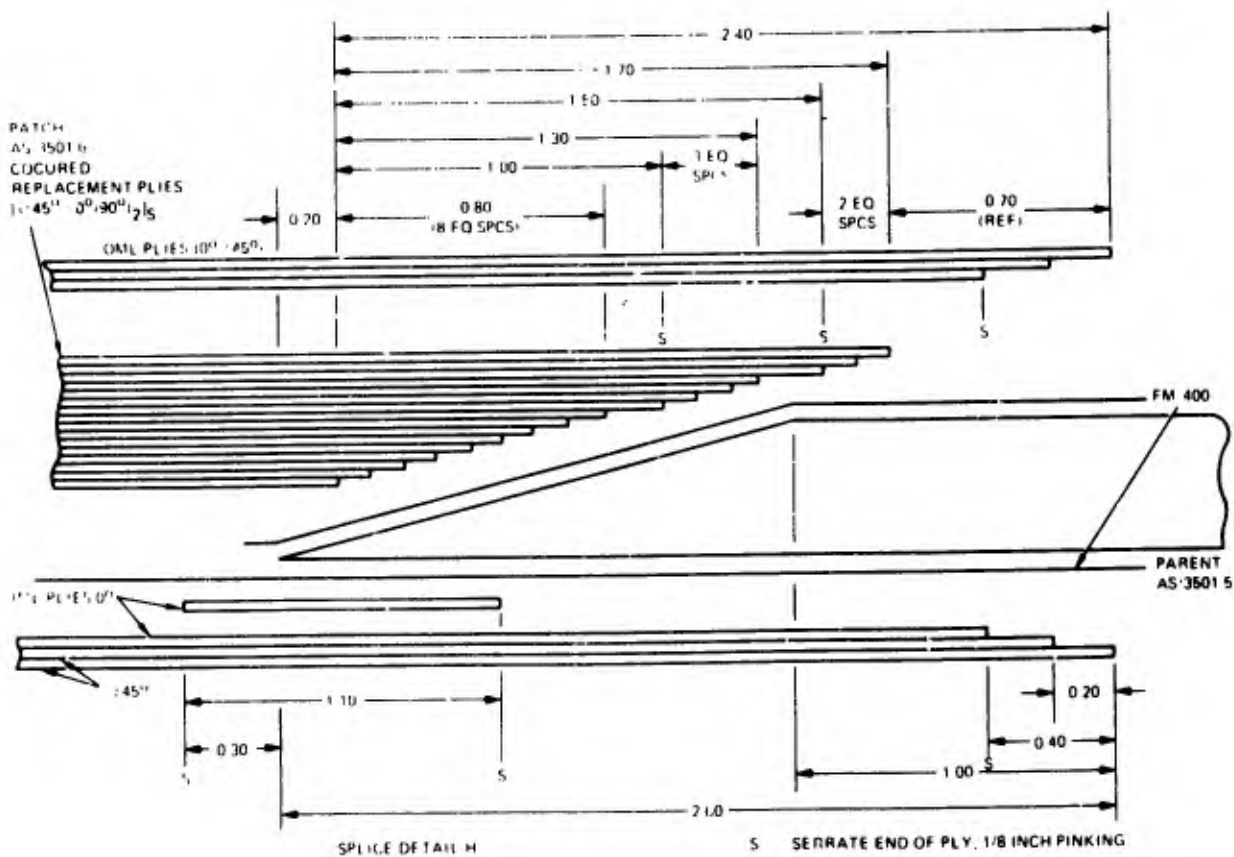


Figure 11. Single Scarf Specimen Type (1) Detail H

83 percent to 97 percent. For test series I and IV, tested on Detail G1 only, failures were tension in the repair plies. Since Detail G2 added strength by lengthening these plies, higher joint efficiencies would be expected approximately in the same ratio as the room temperature results, i.e. 97/83. The joint efficiencies shown in Figure 12 for series I and IV have been increased by this ratio so that all results are on a common basis for a 22-ply patch represented by either Detail G2 or Detail H.

Moisture conditioning of the repair assembly deteriorated the splice for compression results only, where identical specimens were used. Splice failures predominated in the wet tension group also, indicating deterioration of the splice. The prior fatigue loading caused no deterioration in strength. The results plotted are the average of the lower of the two fatigue loaded groups.

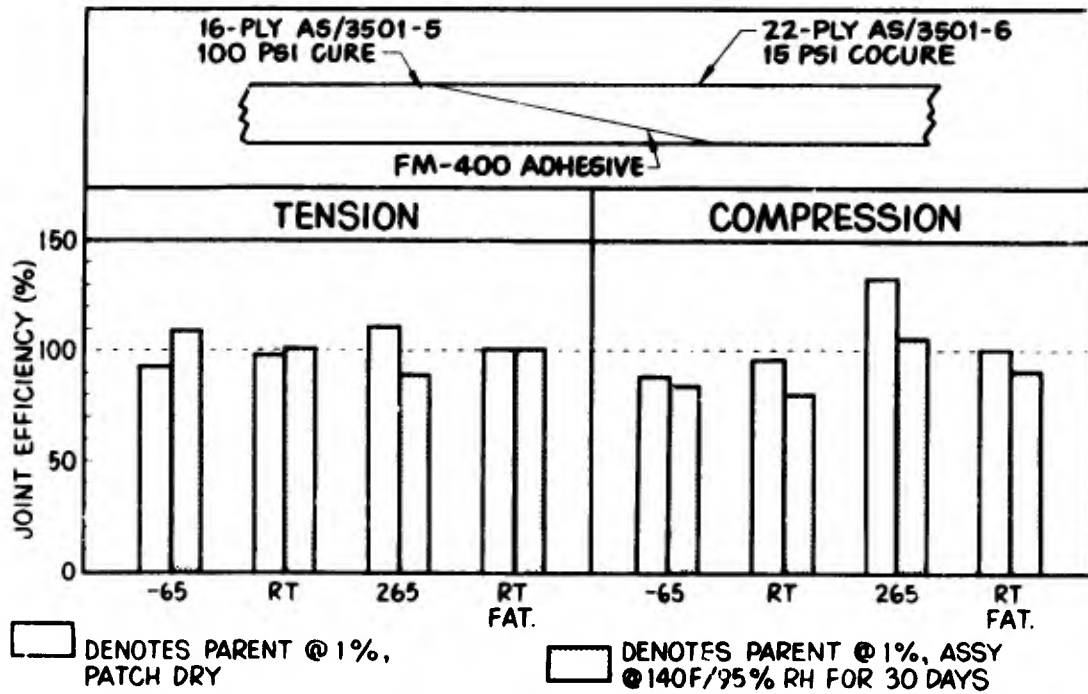


Figure 12. Single Scarf Joint Results

Type 2 — Double Scarf Joint

The double-scarf concept permits a shorter joint length of the same taper ratio with less parent material removed compared to the single scarf. Nine specimens were tested to briefly examine this concept.

Assemblies -201 and -203 shown in Figure 13 describe two cross-sections through the same laminate, involving longitudinal and transverse repairs, respectively. Hence, the stacking orders have the 0-degree and 90-degree plies exchanged in both parent and patch plies for these assemblies (0-degree defined in the loading direction for each specimen). The repair was designed to accommodate maximum load in the longitudinal direction (i. e., the -201 assembly) adding ($\pm 45/0$) plies on both surfaces and serrating the two longest 0-degree plies on either side. The average ultimate tension load for the -201 assembly was 12,380 lb/in, resulting in a joint efficiency of 0.90 as shown in Table 7. All failures were outside the splice, and one failed in the parent laminate. For assembly -203, the average ultimate load shown in Table 7 was 10,910 lb/in., with consistent splice failure.

The parent laminate of the -221 assembly was changed to the stacking order $\left[(\pm 45/0/90)_4 \right]_s$ to better represent design practice applied in production composite components. The length of the two scarfs was changed accordingly to again place their intersection between the pair of ± 45 plies. The stacking order of the repair laminate and other details of the splice were also changed as shown previously in Figure 13. Assembly -221 became the compression face of a sandwich beam for testing, with the results shown in Table 7. The average ultimate load of this configuration was 11,640 lb/in, with the failures consistently in the splice.

The results of the double-scarf specimens are regarded as entirely satisfactory to demonstrate the validity of this approach. Only three plies were added to each side of the laminate in addition to the basic 32 replacement plies. If transverse loads are relatively large, a fourth extra ply in the 90-degree direction should be added to each surface. In the maximum load direction, full strength restoration could probably be attained with one additional ply on the side of the short scarf only, resulting in no additional joint length.

Type 3 — Part-Through Repair of 50-Ply Laminate

To address the case where a scratch, nick, or gouge penetrates the surface of the laminate only part way through, a 50-ply-thick, all graphite/epoxy laminate was

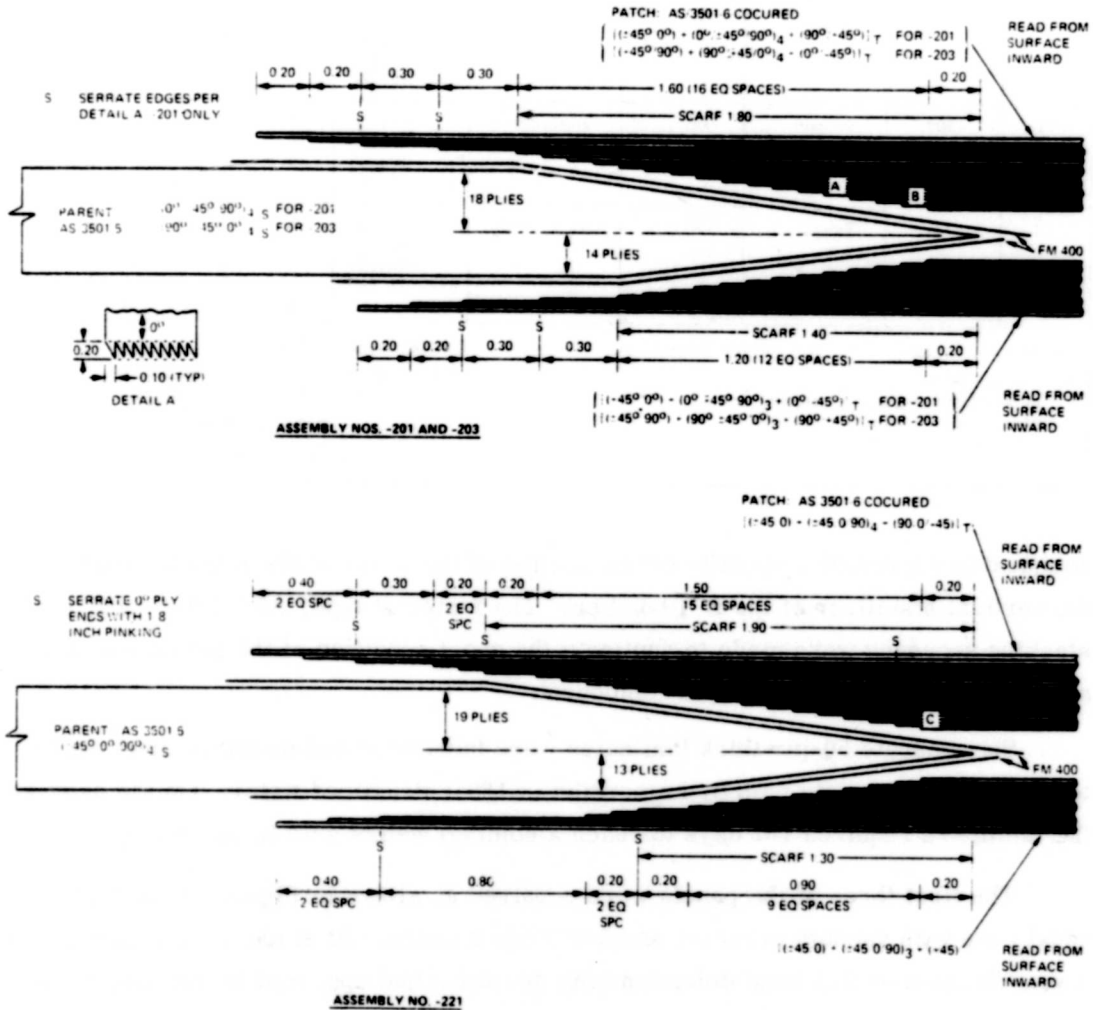


Figure 13. Double-Scarf Repairs, Specimen Type (2)

**TABLE 7. DRY DOUBLE-SCARF REPAIR RESULTS FOR 32-PLY GRAPHITE/
EPOXY LAMINATES, SPECIMEN TYPE (2)**

TEST SERIES	SPLICE DETAIL (a)	TEST CONFIG.	TEST TEMP °F	LOAD SENSE	ULTIMATE LOAD (LB/IN.)	FAILURE MODE (b)	PARENT ALLOWABLE (LB/IN) (c)	JOINT EFF.
11	-201	Coupon	RT	Tension	11,330 13,690 <u>12,110</u> Avg. 12,380	RT PT RT	+ 13,720	0.90
11	-203	Coupon	RT	Tension	10,260 12,420 <u>10,050</u> Avg. 10,910	S S S	+ 13,720	0.80
V1	-221	Beam	RT	Compression	11,300 11,570 <u>12,050</u> Avg. 11,640	S S S	- 15,200	0.77
(a) Reference Figure 7 for splice details.					(c) Reference Table 5			
(b) Failure Modes: RT = Repair laminate tension PT = Parent laminate tension S = Splice failure					PARENT LAMINATES: 0° Longitudinal Spec. No. 201 - [(0/+45/90) ₄] _S Spec. No. 203 - [(90/+45/0) ₄] _S Spec. No. 221 - [(+45/0/90) ₄] _S			

designed to represent a specific cross-section of the cover of the main torque box of the vertical stabilizer of the B-1 bomber. The material type, AS/3501-5, and the stacking sequence were made to duplicate the exact condition of the skin a few inches above the root joint of the actual component.

Six separate 50-ply thick laminates were fabricated and moisture conditioned to a one percent weight gain by immersion in 180°F deionized water. On the average, the laminates required 182 days to reach a nominal weight gain to one percent.

The first three of the panels were repaired as shown in Figure 14, using the usual cure with vacuum pressure and 350°F for 2 hours. After the repair cure cycle, it was discovered that local delaminations (blisters) had appeared in the parent laminates adjacent to the repair in all three panels, varying from one to eight plies deep.

The remaining panels were then dried in an air circulating oven at 180°F. A moisture desorption analysis was made to determine the through-the-thickness moisture profiles, using the procedure of Reference 10. After 16 days of drying, one panel

10. C.H. Shen and G.S. Springer, Moisture Absorption and Desorption of Composite Materials, Journal of Composite Materials, Volume 10, 1976.

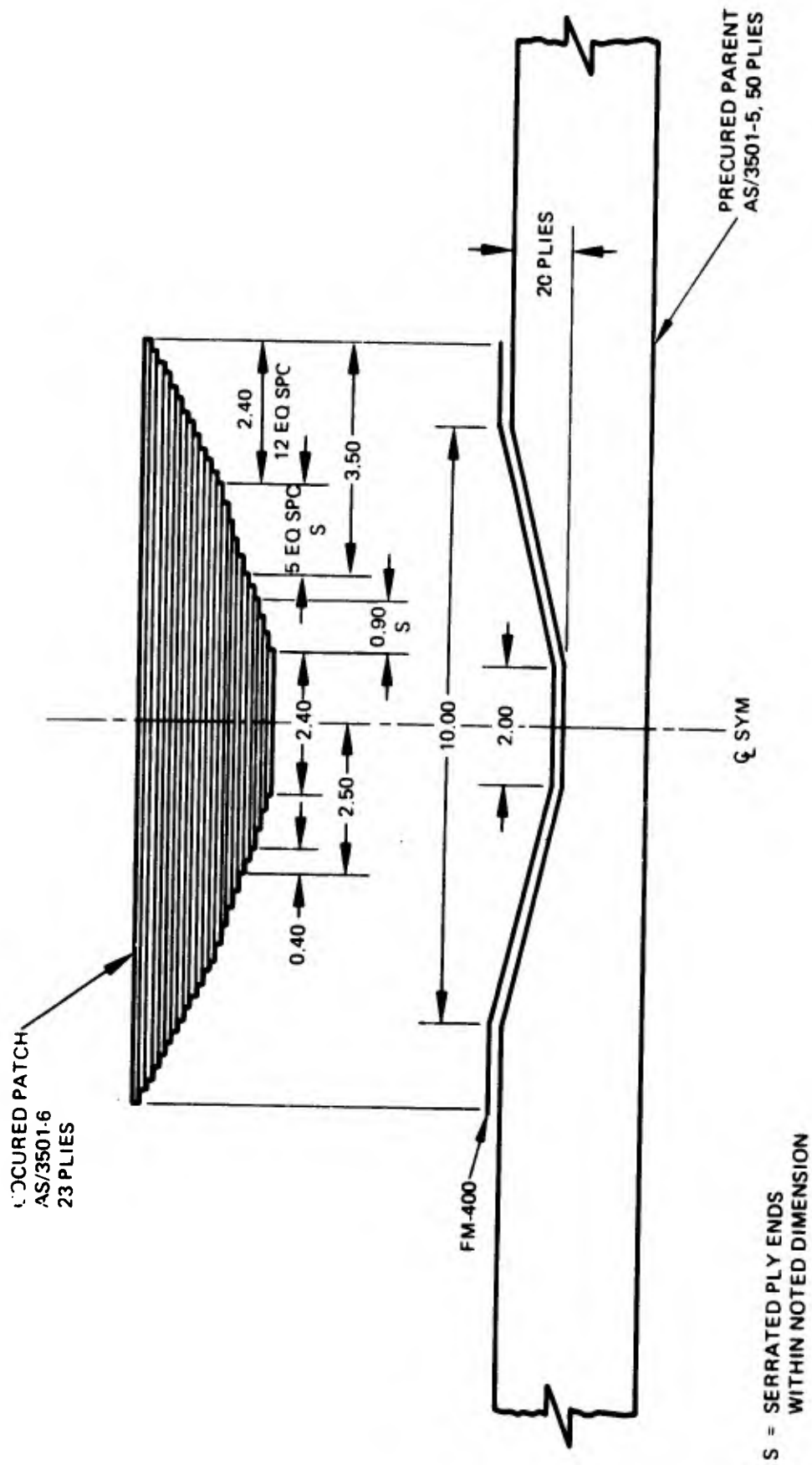


Figure 14. Repair of a Partially-Damaged 50-Ply Graphite Epoxy Laminate
Specimen Type (3)

was repaired using the same cure cycle. Blisters again occurred, but much deeper, near the mid-thickness. The remaining laminates were dried for longer periods, up to 68 days. To assess the possibility of using a lower cure temperature for the repair, panels were fabricated using a 300°F cure temperature and tested for physical and mechanical properties. The properties of the panel cured at 300°F were slightly better than the values for a 350°F cure. As a result, the repairs for the last two panels were cured at 300°F for 4 hours, after drying the parent laminates for up to 68 days. No blisters occurred in these panels.

After all repairs were completed, the 50-inch-long laminates were bonded into sandwich panels which provided four-point beam specimens for structural test. The beam configuration consisted of the 50-ply laminate and a 3/8 inch thick steel slave face on opposite sides of 2-inch deep aluminum honeycomb core. After the sandwich assemblies were bonded, 1.5-inch-wide beam specimens were cut from the wider panels sufficient to fulfill the requirements for Type 3 of the test matrix shown previously in Table 5.

In an attempt to salvage those specimens containing blistered faces (nearly 50% of the total number required), the individual specimens had their local delaminations injected with a low-viscosity 2-part epoxy system routinely employed for such situations requiring elevated temperature integrity. The resin/hardener combination used was Epon 815 with Epon Z hardener in a 20 phr (parts per hundred of resin) mix. The formulation was injected into the delaminations from the exposed edges of the specimens via hypodermic syringe until the voids were completely filled with resin. One edge of the laminate was dammed to prevent loss of resin due to flow during cure and the beams were placed on their sides in an air-circulating oven. The injected resin was then cured at 200° F for one hour followed by 2 hours at 300° F. This cure cycle was chosen to enhance the strength of the resin in those beams to be tested at 265° F.

The moisture condition which caused blistering during these repairs is considered to be a more severe condition than would normally be found in service. Many panels up to 32 plies thick containing 1 percent moisture had been cured previously without blistering. The 50-ply panels which did blister had a very non-uniform moisture distribution through the thickness because of the water immersion method of conditioning which was used. Long time exposure to humidity under actual service conditions would tend to produce a more uniform distribution through the thickness and, except for a few possible worst cases, a lower total moisture content.

Although the tendency to blister is not expected to be a common problem for in-service repairs, it should be recognized as a possibility in extreme conditions. A development program currently in progress ⁽¹¹⁾ may provide a method of determining the moisture content of composite laminates by dielectric measurements.

The details of the moisture absorption and desorption analyses and reasons for the reduced cure temperature were described in the third interim progress report⁽⁶⁾. Test results have not been reported previously and are given in Table 8.

Only three plies ($0^\circ/\pm 45^\circ$) have been added to the patch in addition to the basic replacement plies. All plies were extended across the repair, each at its appropriate length. The parent laminate is scarfed at 0.20-inch/ply and the 0-degree plies of the patch area stepped at 0.30 inch/ply. Load transfer analysis determined that one of these 0.30-inch steps (the one at the end of the shortest patch ply) would transfer 4.52 percent of the longitudinal load. Another 0.30 inch step (splicing the outer group of damaged 0-degree plies) would transfer 4.13 percent of the load.

Three new variables have been introduced into the Type 3 specimen: (a) the repair adhesive, (b) panel drying time and (c) the existence of a blister repair. Some of the repairs were bonded with FM-400 adhesive and some with RB-398 adhesive which have similar properties. While average values are shown for specimens repaired with a given adhesive and tested at the same temperature, data within a group are often not consistent due to the other variables. Both adhesives appear to perform well if the panel was adequately dried before the repair was made. A drying time of 16 days appears adequate insofar as the strength values are concerned, though not sufficient to prevent blistering during the repair cure at 350°F. Blister repairs appear to have been adequately made. The results display no consistent strength deterioration due to blister repairs.

Attempting to repair the laminate with 1 percent average moisture content (0 days drying) appears to have been the only feature of the repair technique which consistently resulted in reduced strength, as shown in Table 8.

In tension, if the laminate was adequately dried before the repair, the failure mode was generally a tension failure through the repair which involved the splice at 265°F. The splice failures with relatively poor results at -65°F are attributed to the moisture condition of the laminate being repaired, not to the type of adhesive used.

11. A.O. Kays, Determination of Moisture Content in Composites by Dielectric Measurements, Contract No. F33615-78-C-3216, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, 45433 (Lockheed-Georgia Co.)

TABLE 8. TYPE 3 - 50-PLY PANEL REPAIR RESULTS

(a) TEST SERIES	REPAIR ADHESIVE	TEST TEMP (°F)	FATIGUE MAX LOAD (b) (LB/IN)	LOAD SENSE	N _x = ULTIMATE LOADING (LB/IN)	(c) FAILURE MODE	(d) PANEL DRIED (DAYS)	BLISTER REPAIR	(e) TENSION ALLOWABLE (LB/IN)	(f) REPAIR EFF
CONTROL SPECIMENS - NO DAMAGE										
		RT	None	Tension	32,060	PT	68	No	18,270	
		RT	None	Compr.	-29,300	PC	68	No	24,460	
REPAIR SPECIMENS										
I	FM400	-65	None	Tension	14,970	SP	0	Yes		
I	FM400	-65	None	Tension	14,620	SP	0	No		
					Avg. = 14,790				16,540	0.89
I	RB398	-65			20,980	CB	68	No		
I	RB398	-65			23,970	RT, SP	16	Yes		
I	RB398	-65			28,420	RT	16	Yes		
					Avg. = 24,460				16,540	1.48
II	FM400	RT			23,360	RT	0	No		
II	FM400	RT			25,360	CB	68	No		
II	FM400	RT			17,710	CB	0	No		
					Avg. = 22,140				18,270	1.21
II	RB398	RT			28,670	RT	68	No		
II	RB398	RT			24,540	RT	16	Yes		
II	RB398	RT			27,950	RT	16	No		
					Avg. = 27,050				18,270	1.48
III	FM400	265			12,790	SP	0	Yes		
III	FM400	265			20,300	RT, SP	68	No		
III	FM400	265			13,930	RT, SP	0	No		
					Avg. = 15,670				15,310	1.02
III	RB398	265			14,820	CB	68	No		
III	RB398	265			18,330	RT, SP	68	No		
III	RB398	265	None		21,810	RT, SP	68	No		
					Avg. = 18,320				15,310	1.20
IV	FM400	RT	5525		12,710	SP	0	Yes		
IV	FM400	RT	5525		15,720	SP	0	No		
IV	FM400	RT	5525		17,970	SP	0	Yes		
					Avg. = 15,470				18,270	0.85
IV	RB398	RT	5525		24,150	RT, SP	16	Yes		
IV	RB398	RT	5525		23,280	RT, SP	16	Yes		
IV	RB398	RT	5525	Tension	27,660	CB	68	No		
					Avg. = 25,030				18,270	1.37
V	FM400	-65	None	Compr.	-22,410	SP	0	No		
V	FM400	-65			-25,180	SP	0	No		
					Avg. = -23,790				31,350	0.76
V	RB398	-65			-31,040	PC	68	No		
V	RB398	-65			-30,740	CB, II	68	No		
V	RB398	-65			-30,010	PC	68	No		
					Avg. = -30,600				31,350	0.98

(FOOTNOTES ON LATER PAGE)

TABLE 8. TYPE 3 - 50-PLY PANEL REPAIR RESULTS (Continued)

(a) TEST SERIES	REPAIR ADHESIVE	TEST TEMP (°F)	FATIGUE MAX LOAD (b) (LB/IN)	LOAD SENSE	N _x = ULTIMATE LOADING (LB/IN)	(c) FAILURE MODE	(d) PANEL DRIED (DAYS)	BLISTER REPAIR	(e) (f) TENSION ALLOWABLE (LB/IN)	REPAIR EFF
VI	FM400	RT	None	Compr.	-16,750	SP	0	Yes		
VI	FM400	RT			-28,130	RC	0	No		
VI	FM400	RT			***	IL, NO	0	No		
					Avg. = -22,440				24,460	0.92
VI	RB398	RT			-25,020	RC	64	No		
VI	RB398	RT			-24,940	RC	64	No		
VI	RB398	RT			-20,900	RC	64	No		
					Avg. = -23,620				24,460	0.97
VII	FM400	265			-11,970	SP	0	Yes		
VII	FM400	265			-18,830	RC, SP	68	No		
VII	FM400	265			-15,390	RC, SP	0	Yes		
					Avg. = -15,400				15,160	1.02
VII	RB398	265			-18,000	SB	68	No		
VII	RB398	265			-17,920	SB	68	No		
VII	RB398	265	None		-15,970	SB	68	No		
					Avg. = -17,300				15,160	1.14
VIII	FM400	RT	5525		-13,430	SP	0	Yes		
VIII	FM400	RT	5525		-16,300	SP	0	Yes		
VIII	FM400	RT	5525		-17,410	SP	0	Yes		
					Avg. = -15,710				-24,460	0.64
VIII	RB398	RT	5525		-29,290	RC	68	No		
VIII	RB398	RT	5525		-30,070	PC	68	No		
VIII	RB398	RT	5525	Compr.	-18,100	CB	68	No		
					Avg. = -25,820				-24,460	1.06
XI	FM400	265	None	Tension	11,030	SP	0	Yes		
XI	FM400	265			20,570	SP	68	No		
XI	FM400	265			14,080	SP	0	Yes		
					Avg. = 15,230				15,310	0.99
XI	RB398	265			15,990	RT, SP	68	No		
XI	RB398	265			13,660	SP	68	No		
XI	RB398	265		Tension	17,020	CB	16	No		
					Avg. = 15,560				15,310	1.02
XV	FM400	265		Compr.	***	IL, NO	0	Yes		
XV	FM400	265			13,110	SP	68	No		
XV	FM400	265			14,070	RC, SP	0	Yes		
					Avg. = 13,590				15,160	0.90
XV	RB398	265			12,640	SP	68	Yes		
XV	RB398	265			16,310	SP	64	No		
XV	RB398	265	None	Compr.	13,620	SP	16	No		
					Avg. = 14,190				15,160	0.94

(FOOTNOTES ON NEXT PAGE)

TABLE 8. TYPE 3 - 50-PLY PANEL REPAIR RESULTS (Concluded)

NOTES:

- (a) AS/3501-5 Laminate ($\underline{+45/90}_2/\overline{+45}/\underline{+45/0}_4/\overline{+45}/\underline{+45/0}_3/\overline{+45}/\underline{+45/0}_2$)_s
- (b) F-16 vertical stabilizer fatigue spectrum (fully reversible) with maximum $N_x = 5525$. Lb/In for two lifetimes.
- (c) Failure Modes:
- CB = Core Bond
 - IL = Interlaminar Shear
 - NO = No Repair Failure
 - PC = Parent Compression
 - PT = Parent Tension
 - RC = Repair Compression
 - RT = Repair Tension
 - SB = Splice Buckling
 - SP = Splice
- (d) Panels dried 0 or 16 days were (repair) cured at 350°F. Panels dried 64 or 68 days were (repair) cured at 300°F.
- (e) RT Compression Ultimate Allowable = 0.80 (116,500 psi x 50 plies x 0.00525 inch/ply) = 24,460 lb/inch.
Average strength data taken from Table 3-II of Reference 12. Minimum of six tests conducted was 90% of average. Allowables at other temperatures scaled in proportion to the values in Table .
- (f) RT Tension Ultimate Allowable = 0.80 (87,010 psi x 50 plies x 0.00525 inch/ply) = 18,270 lb/ inch.
Average strength data taken from Table 3-I of Reference 12. Minimum of six tests conducted was 81% of average. Allowables at other temperatures scaled in proportion to the values in Table 1.

*** Failure mode indicates the test is not valid.

Similar results are observed in compression; splice failures and lower ultimate loading is associated with no panel drying. At -65°F two specimens failed in the parent material in compression.

Evaluation of the Type 3 repair is complicated by the additional variables involved. In comparison to the allowables determined from Reference 12, the repair design is entirely satisfactory. Failure modes and ultimate loading results suggest that two or three additional 0-degree plies would improve the repair to approximately the same strength as the undamaged control specimen. Such plies should be buried within the outer eight plies and should be serrated at the ply ends.

For specimens with fatigue loading (Series IV and VIII), a fully reversible vertical stabilizer fatigue spectrum was imposed for two airplane lifetimes with the maximum loading $N_x = 5525 \text{ Lb/In.}$ Compression loads were not truncated in this spectrum, hence the maximum loading is 30.2 percent of the parent tension ultimate allowable and 22.6 percent of the parent compression ultimate allowable. Comparison of the two groups tested in tension shows a significant improvement for those laminates which were dried before the repair was made. The type of adhesive used is not considered significant based on the prior experience with FM-400. Comparison of the two groups tested in compression leads to the same observation. Comparison of the repair efficiencies of those beams with prior fatigue history to those without indicates that residual strength was reduced on those repairs where the parent laminate was wet (average moisture content = 1 percent) but the strength was not reduced if the laminate was adequately dried. The failure modes also indicate the importance of drying the laminate before bonding the repair when maximum strength is required.

The results of static tests at 265°F on wet 50-ply Type 3 specimens (Test Series XI and XV) also show the importance of bonding to a dry surface. Overall, the wet (i. e., moisture conditioned after repair) specimens yield somewhat lower repair efficiencies than those which remain dry.

Type 4 — Part-Through Repair of 64-Ply Hybrid Laminate

As with the 50-ply graphite/epoxy laminate repair just discussed, a second B-1 empennage component was addressed relative to partial thickness damage removal and

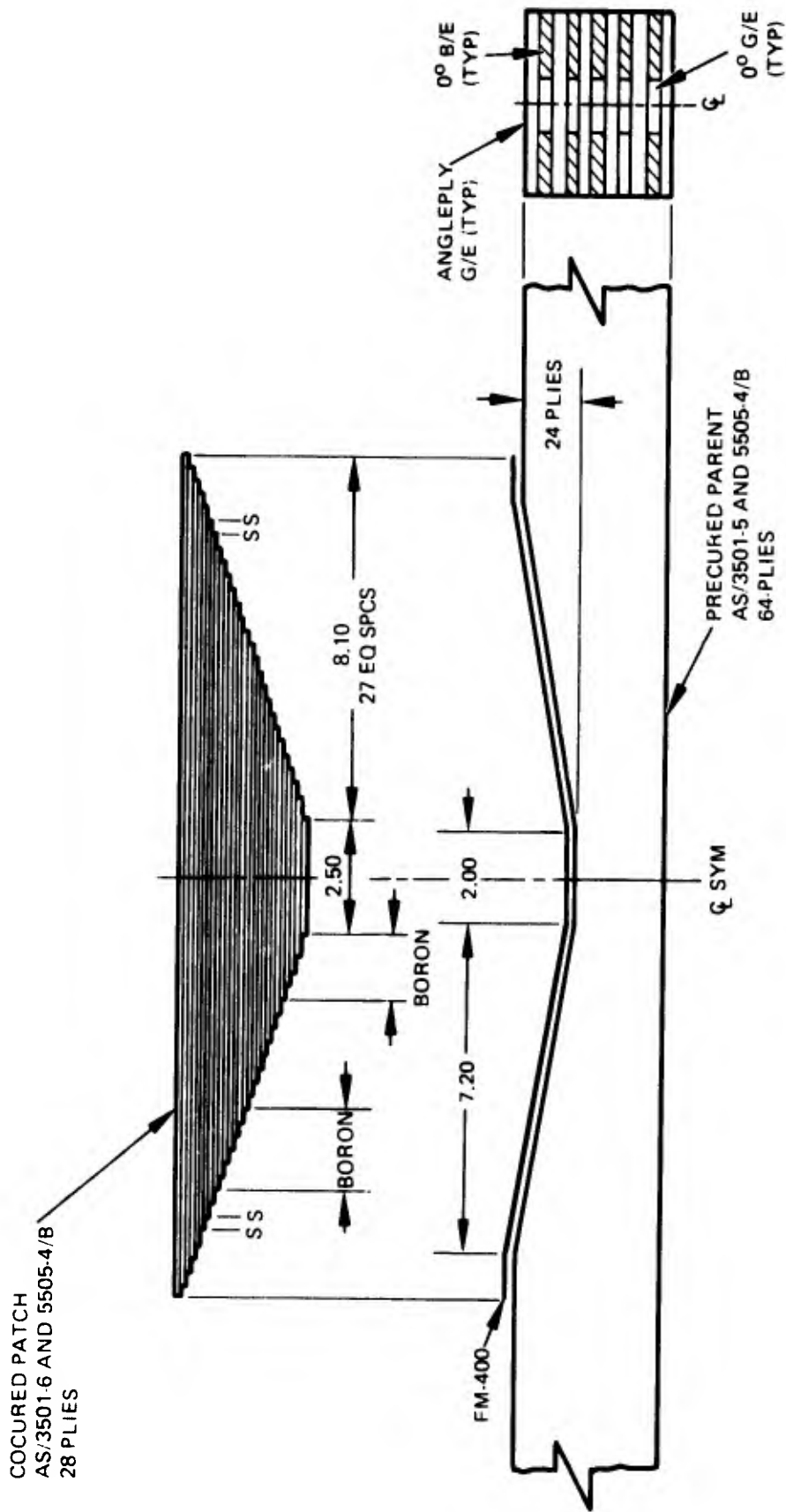
-
12. D.E. Parker, Development of a Low Cost Composite Vertical Stabilizer, Contract No. F33615-74-C-5164, Second Quarterly Progress Report, Rockwell International, NA-75-99-2.

repair. This second component, representative of the B-1 graphite-boron/epoxy cover of the horizontal stabilizer just outboard of station 105.90, was processed similarly to 50-ply Type 3 laminates, with the exclusion of the moisture conditioning. There are two reasons why the effects of parent laminate moisture conditioning were not considered: 1) the time required to achieve a moisture content of 1 percent in this 64-ply thick laminate, nominally 16 months, was considered impractical for this program; and 2) the effects of moisture and the environment on laminates of this very type were being assessed by Grumman under their work for the Air Force ⁽¹³⁾ on the development of the composite horizontal stabilizer.

Parent laminate fabrication was more elaborate than it had been for an all-graphite panel in that the cross-section of the cover of the B-1 horizontal tail consists of grouped strips of boron/epoxy plies over spar caps separated by continuous books of graphite/epoxy. In addition, the boron/epoxy plies were interrupted by narrow strips of graphite/epoxy which acted as drilling strips along the spar caps of the tail where the covers were mechanically attached with countersunk fasteners. The cross-section of the cover over the spars, and hence the cross-section of the specimens being fabricated for investigation within the LACOSR program, is presented in Figure 15 along with a schematic of the repair concept for this laminate.

The repair reflects a symmetrical, partial-thickness damage-removal profile and patch detail. The 24 plies of graphite and boron removed were replaced by the same materials and orientations, plus four additional graphite/epoxy plies, to enhance load transfer across this particularly highly loaded laminate. All parent 64-ply hybrid panels were machined to the contour shown in Figure 15. Rough machining of the scarfed area was done using a fiberglass-reinforced aluminum oxide grinding wheel which is very inexpensive and can be discarded when worn. Final machining to the desired profile was accomplished with the more expensive diamond-coated cup-type cutters. With this two-stage process, the desired cutout dimensions were obtained with minimum tool expense and machining time by avoiding resintering of the diamond tools. After machining, the patch plies were cocured with Reliabond 398 adhesive into the scarfed pocket using only vacuum pressure.

13. ———, Advanced Development of Conceptual Hardware, Air Force Materials Laboratory, Wright-Patterson AFB, Ohio, 45433, Contract No. F33611-73-C-5173, (Grumman Aerospace Corp.)



S = SERRATED PLY END

Figure 15. Repair of a Partially Damaged 64-Ply Hybrid Laminated Specimen Type (4)

After material removal and repair processing, all laminates were bonded into sandwich panels for test purposes. As for the 50-ply specimens, the 64-ply, 70-inch-long beams consisted of the composite face and a steel slave face on either side of 2-inch-deep aluminum core. Each specimen was 2-inches wide and was loaded as a four-point beam to the requirements of Table 5. The fatigue spectrum used for the cyclically loaded specimens is a true random flight-by-flight spectrum for the horizontal stabilizer of the B-1 obtained from the AFFDL for this purpose.

Three undamaged beams were tested (2 tension, 1 compression) with identical laminate configuration as the repaired beams. While the number of tests involved was small, the size of the test specimen was much larger than usual, i. e., in each beam, a 2 x 20 inch area of laminate was fully loaded as compared to the usual 0.5 x 2 inch tension specimen. Hence, each of these control specimens was reasonably representative of the parent laminate, and the two tension control specimens gave almost identical results. To define an allowable, of course, some reduction should be made from these typical results. However, such a reduction, with the information available, would be quite arbitrary. Hence, the repair results are here compared to the control test results with no reduction. At temperatures other than room temperature, the control test results are scaled in proportion to the values for the all-graphite/epoxy quasi-isotropic laminate in Table 1.

Test results are tabulated in Table 9. At all temperatures, the predominate static failure mode is tension failure in the repaired area. At 265°F, some indication of splice failure is seen.

Six of the repaired beams were subjected to two lifetimes of fatigue loading using the B-1 horizontal stabilizer fatigue spectrum set at nominal maximum and minimum skin loadings of 16600 and -17400 lb/in, respectively. Four of the six beams survived the two lifetimes of fatigue loading with no visible deterioration. One showed a small delamination at approximately one lifetime which grew to about 1.5 inches and stopped growing. On one specimen, a delamination 1.25 inches long was noted about 1.8 inch below the surface before loading began. This delamination grew rapidly to 16 inches at 0.1 of the first lifetime of fatigue loading. The delaminated material buckled under each compression loading, soon becoming useless. However, the piece did remain attached and the specimen completed the two lifetimes of fatigue.

In residual strength testing, the three specimens loaded in tension all failed in the repair splice, then failed through the net section of the parent laminate. The average repair (or ultimate) failure load was 89 percent of the parent reference strength.

TABLE 9. TYPE 4 - HYBRID PANEL REPAIR RESULTS

TEST SERIES	TEST TEMP (°F)	TWO LIFETIMES(a) FATIGUE LOADS		LOADING SENSE	LOAD AT FAILURE (LB/IN)	(b) FAILURE MODE	PARENT (c) REFERENCE STRENGTH (LB/IN)	REPAIR EFF
		N _X MAX (LB/IN)	N _X MIN (LB/IN)					
CONTROL SPECIMENS, NO DAMAGE								
	RT	None	None	Tension	31,009	PT		
	RT	None	None	Tension	31,319	PT		
					Avg. = 31,160			
	RT	None	None	Comp.	-42,110	IL		
24-PLY DAMAGE REPAIRED SPECIMENS								
	-65	None	None	Tension	34,182	RT		
I	-65	None	None	Tension	27,287	RT		
I	-65	None	None	Tension	31,960	RT		
					Avg. = 31,140		28,210	1.10
II	RT	None	None	Tension	33,975	RT		
II	RT	None	None	Tension	33,182	RT		
II	RT	None	None	Tension	33,189	RT		
					Avg. = 33,450		31,160	1.07
III	265	None	None	Tension	18,434	SP		
III	265	None	None	Tension	27,297	RT, SP		
III	265	None	None	Tension	30,246	RT		
					Avg. = 25,330		26,120	0.97
TENSION RESIDUAL STRENGTH SPECIMENS								
IV	RT	16600	-17400	Tension	26,790	SP		
IV	RT	16600	-17400	Tension	26,310	SP		
IV	RT	16600	-17400	Tension	30,070			
					Avg. = 27,720		31,160	0.89
COMPRESSION RESIDUAL STRENGTH SPECIMENS								
VIII	RT	16600	-17400	Comp.	-41,190	SP		
					(-22220)*			
VIII	RT	16600	-17400	Comp.	-35,820	SP, F		
					(-22770)*			
VIII	RT	16600	-17400	Comp.	-35,130	SP, FF		
					(**)			
					Avg. -37,380		-42,110	0.89
					(-22500)*		-42,100	(0.53)*

$[(+45/90/\pm 45)_G / (0_6)_B / (+45_4)_G / (0_5)_B / \pm 45_2/0)_G / (0_3)_B]_S$ Boron and graphite hybrid Laminate 0.70 inch each side. In the center 0.60 inch strip the boron is replaced with graphite.

**Failure during spectrum loading. *Repair failure, parent material intact.

- (a) B-1 horizontal stabilizer fatigue spectrum.
- (b) Failure modes: SP= Splice, F= 1.5 inch fatigue delamination, FF= Complete fatigue delamination of most of the patch material, PT= Parent Tension, IL= Interlaminar Shear, RT= Repair Tension
- (c) Parent Reference Strength at room temperature = average of the tension control specimens. Parent Reference Strength at other temperatures scaled in proportion to the values in Table 1.

Those tested in compression failed first in the splice, with all or most of the patch breaking loose, followed by ultimate failure at a much higher load. The specimen which had lost the effectiveness of a large part of the patch during fatigue loading suffered no further failure of the repair until ultimate failure load was reached. Evaluating the repair failure compression data becomes somewhat ambiguous and this specimen has not been included in the repair failure average. The repair failures of the hybrid laminate in compression both during and after the fatigue testing were the least successful results in the program. However, even these results indicate that if the repair is soundly bonded, it would be adequate for the fatigue loads and the ultimate design loads of the mechanically attached panel since these would be much less than the parent reference strength.

Type 5, 6, 7 - Blind Side Single Scarf Repairs

The configuration used for repair of a typical 16 ply monolithic laminate with access from one side only is shown in Figure 16. As previously discussed in Paragraph 2.3, the strength is increased significantly by having one or more plies overlapping the end of the scarf joint on the inner surface. For the blind repair, this was provided by a 6-ply precured doubler which, after bonding in place, also gave a surface on which the replacement plies could be placed and cured.

A precured AS/3501-6 laminate ($\pm 45/0$)_s was chosen for the inside patch (doubler) because of the fact that: (1) no new bonding materials or logistics are involved, (2) tapering of the patch is made easy by cutting the plies to the proper size and serrating the edge where necessary, and (3) forming the patch to a compound curvature, if necessary, can be done without expensive tools, using the outer surface of the part to make a mold. The laminate stacking order is symmetric and continues to be so even as internal plies are terminated.

Locating the internal patch may require some ingenuity. With access limited to only the external surface, the panel must be installed through the damage cutout. A long narrow hole offers the easy solution of rotating the panel into place; however, careful bending of the 6-ply panel to insert it through a circular hole could be considered. The six-ply panel can be elastically bent to a 2-inch radius of curvature without damage. Once in place, the 6-ply panel is held against the blind (inner) surface of the panel while holes are drilled through both pieces from the outside and Clecos are installed. A method of doing this on a large realistic panel is described in Paragraph 4.3 for a large scale test panel.

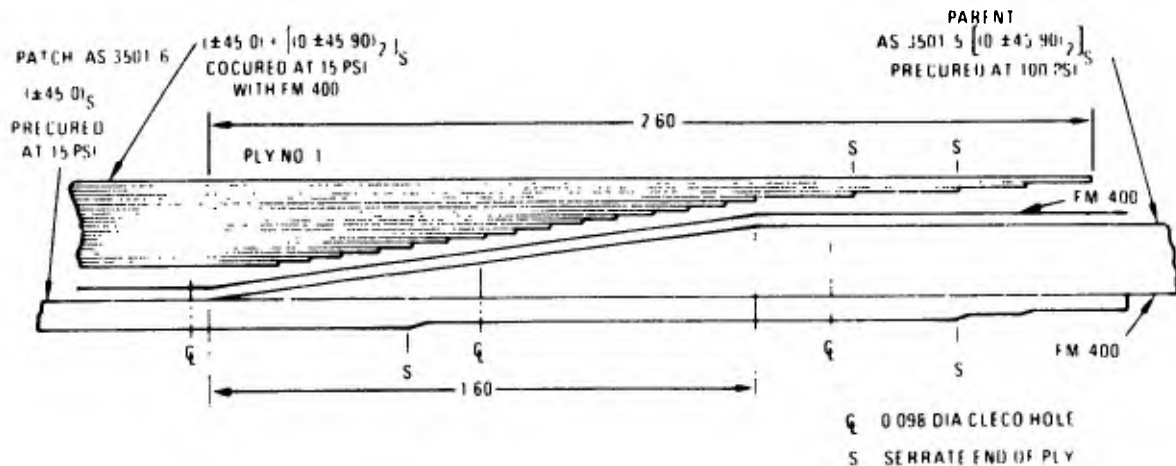


Figure 16. Blind-side Repair Specimen Type (5)

The drill template plate remained in place during the bonding cure to provide support to the Clecos beyond the edge of the parent laminate and through the scarfed areas as shown schematically in Figure 16. The holes must be filled after removal of the Clecos to prevent vacuum leaks during the secondary curing of the flush patch. The cured part of the repair, consisting of (+45/0) plies plus replacement plies, is bonded in the second cure with vacuum pressure, bagged on the outside surface only.

Besides Cleco bonding of the blind side panel, two other methods were attempted using an otherwise identical splice. These two alternate methods consisted of applying the bonding pressure for attachment of the internal patch by vacuum pressure only, (Type 7) and a combination of Clecos and blind rivets (Type 6). The results of these tests indicate that the three methods produced essentially the same joint strength. Hence, Cleco bonding was used on all subsequent tests of the blind-side repair of sandwich panels.

Detailed test results of the blind-side repair tests at various temperature, moisture, and load conditions are given in the second interim progress report⁽⁵⁾, and are summarized in Figure 17. The lowest joint efficiency shown is 84 percent and, in general, the results indicate that the blind-side repair can be made as good as one with access to both sides if the blind-side patch can be adequately bonded using techniques like those investigated herein.

Type 12 - Single Scarf Repair of T300/5208 Material

The basic program was expanded to include the repair of a monolithic parent laminate of T300/5208 graphite/epoxy which closely resembled those used in the empennage of the F-16 fighter. The specific objective was to determine whether or not

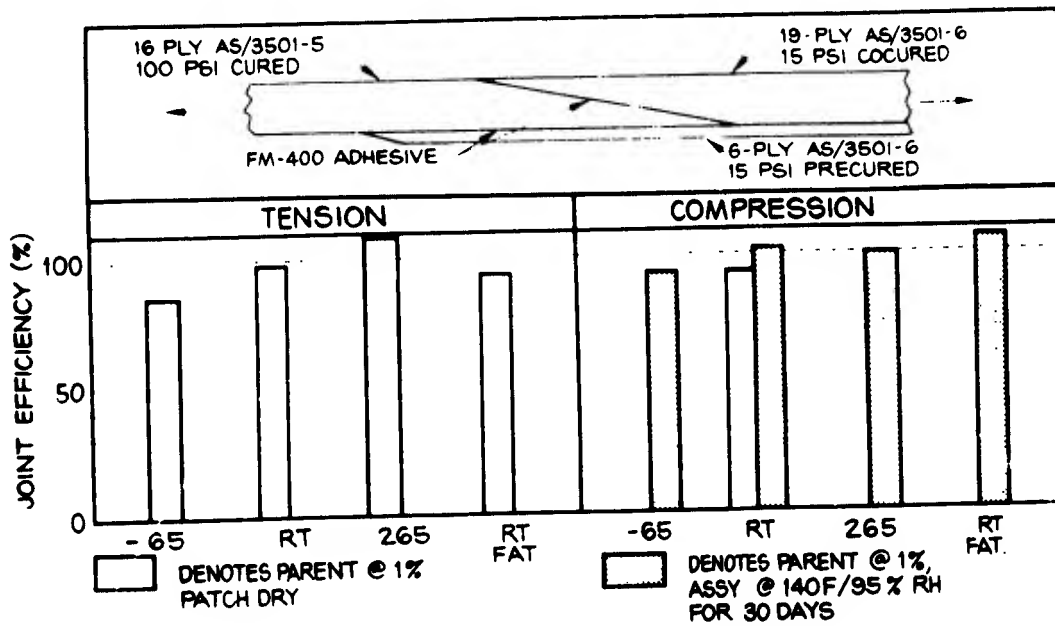


Figure 17. Blind Monolithic Joint Results

using the AS/3501-6 patch material and processing to repair the F-16 parent material provided as efficient a repair joint as it had when repairing AS/3501-5 graphite/epoxy parent material. The patch details and tests specimen configurations to be used were identical to Type 1 previously described.

All of the 16-ply parent laminates were moisture conditioned to the desired 1 percent weight gain by immersion in 180°F deionized water. After conditioning, the laminates were scarfed to the same 18:1 taper which was used for all previous single- and double-scarf joint concepts investigated.

The parent laminates were repaired using AS/3501-6 graphite/epoxy and Relia-bond 398 adhesive co-cured under vacuum pressure at 350F for 2 hours. After repair, those specimens so designated were exposed to 30 days of 140°F/95% relative humidity conditioning prior to test. Fatigue loading used the same F-5E fatigue spectrum as was used in the testing of the 16-ply AS/3501-5 Type 1 specimens previously.

Mechanical property tests on the T300/5208 parent graphite/epoxy are presented in Table 10. These data were used to assess repair joint efficiencies by comparing the failure load of the repaired coupons and beams (per inch of specimen width) to the respective tension or compression allowable. The transverse tensile strain allowable was used to define the magnitude of the peak load in the F-5E fatigue spectrum to be used for cyclic loading of Series IV and VIII specimens prior to residual strength test-

TABLE 10. PARENT LAMINATE ALLOWABLES FOR
T300/5208 WITH 1% MOISTURE CONTENT

PARAMETER	LAMINATE TESTED	ALLOWABLE AT TEMPERATURE		
		-65F	RT	265F
90° Tensile Strain, μ in/in.	$(0_{16})_T$	--	4580	--
0° Tensile Strength, lb/in.	$(\pm 45/0/90)_S$	4425	5185	4770
0° Compressive Strength, lb/in.	$[(\pm 45/0/90)_2]_S$	7190	6495	3870

Laminates cured per General Dynamics specification FPS-2021C, 11 September 1975. Allowables quoted are 80% of average test results.

ing. Comparison of the T300/5208 allowables with comparable values for the AS/3501-5 material in Table 1 shows significantly higher values for the AS/3501-5 material.

Test results are tabulated in Table 11. For "dry" static tension coupons (parent laminate at 1 percent moisture with no reexposure after the repair), those tested at -65°F failed in parent material tension while at 265°F the typical failure mode was in the splice. At all temperatures the ultimate load is typical of the parent material.

Compression tests on dry repairs were made using beam specimens with the laminate in compression (Test Series V, VI and VII). As shown by the joint efficiencies, this joint concept restored the full typical compression ultimate strength of the parent laminate of each test temperature. The failure modes indicate that the splice tends to become critical with increased temperature, though the performance of the splice relative to the parent allowable remains excellent.

Static tests at 265°F were made on wet repairs, i.e., reexposed after the repair was made, Test Series XI and XV. All specimens failed in the splice as would be expected from prior results. The joint efficiency of Test Series XI (tension at 265°F) of 90 percent is the lowest of the Type 12 specimens, and is the only value less than 100 percent.

Fatigue loading was applied to Series IV and VIII with no significant effect on the repair joint strengths. There was a small difference in the maximum fatigue loading because the tension coupons were loaded to the same loading as the Type 1 specimens while the compression beams were loaded to the same strain as the Type 1 specimens. Given the elastic modulus of 7.73 msi for the T300/5208 parent laminates

TABLE 11. TYPE 12 - SINGLE SCARF REPAIR RESULTS ON T300/5208

TEST SERIES	TEST TEMP (°F)	TWO LIFETIMES FATIGUE LOADS (a)		LOADING SENSE	ULTIMATE LOADING (LB/IN)	FAILURE MODE (b)	PARENT ALLOWABLE (LB/IN) (c)	JOINT EFF
		MAX LOAD (LB/IN)	MAX STRAIN (µIN/IN)					
I	-65	None	None	Tension (Coupon)	5715	PT	4425	1.22
I	-65	↓	↓		5305	PT		
I	-65				5180	PT		
					Avg. = 5400			
II	RT	↓	↓	↓	6667	SP	5185	1.24
II	RT				6014	SP		
II	RT				6601	PT		
					Avg. = 6430			
III	265	↓	↓	↓	5814	SP	4770	1.19
III	265				5694	SP		
III	265				5530	SP		
					Avg. = 5680			
IV	RT	2820	4180	↓	5722	SP	5185	1.08
IV	RT	2820	4180		5315	SP		
IV	RT	2820	4180		5774	SP		
					Avg. = 5604			
V	-65	None	None	Compr. (Beam)	7997	PC	7190	1.19
V	-65	↓	↓		8701	PC		
V	-65				9023	PC		
					Avg. = 8574			
VI	RT	↓	↓	↓	7535	RC	6495	1.24
VI	RT				8012	PC		
VI	RT				8634	RC		
					Avg. = 8060			
VII	265	↓	↓	↓	5151	SP	3870	1.29
VII	265				4518	SP		
VII	265				5276	SP		
					Avg. = 4982			
VIII	RT	3084	4570	↓	8504	RC	6495	1.40
VIII	RT	3084	4570		10001	RC		
VIII	RT	3084	4570		8791	RC		
					Avg. = 9009			
XI	265	None	None	Tension (Coupon)	4028	SP	4770	0.90
XI	265	↓	↓		4447	SP		
XI	265				4392	SP		
					Avg. = 4289			
XV	265	None	None	Compr. (Beam)	4698	SP	3870	1.26
XV	265				4580	SP		
XV	265				5340	SP		
					Avg. = 4873			

T300/5208 Parent laminate $[(+45/0/90)_2]_s$, RB398 Adhesive, AS/3501-6 Patch material

- (a) Tension spectrum fatigue, F-5E lower wing skin root.
- (b) Failure Modes: PC= Parent Compression, RC= Repair Compression, SP= Splice
- (c) Parent laminate strength per Table 10.

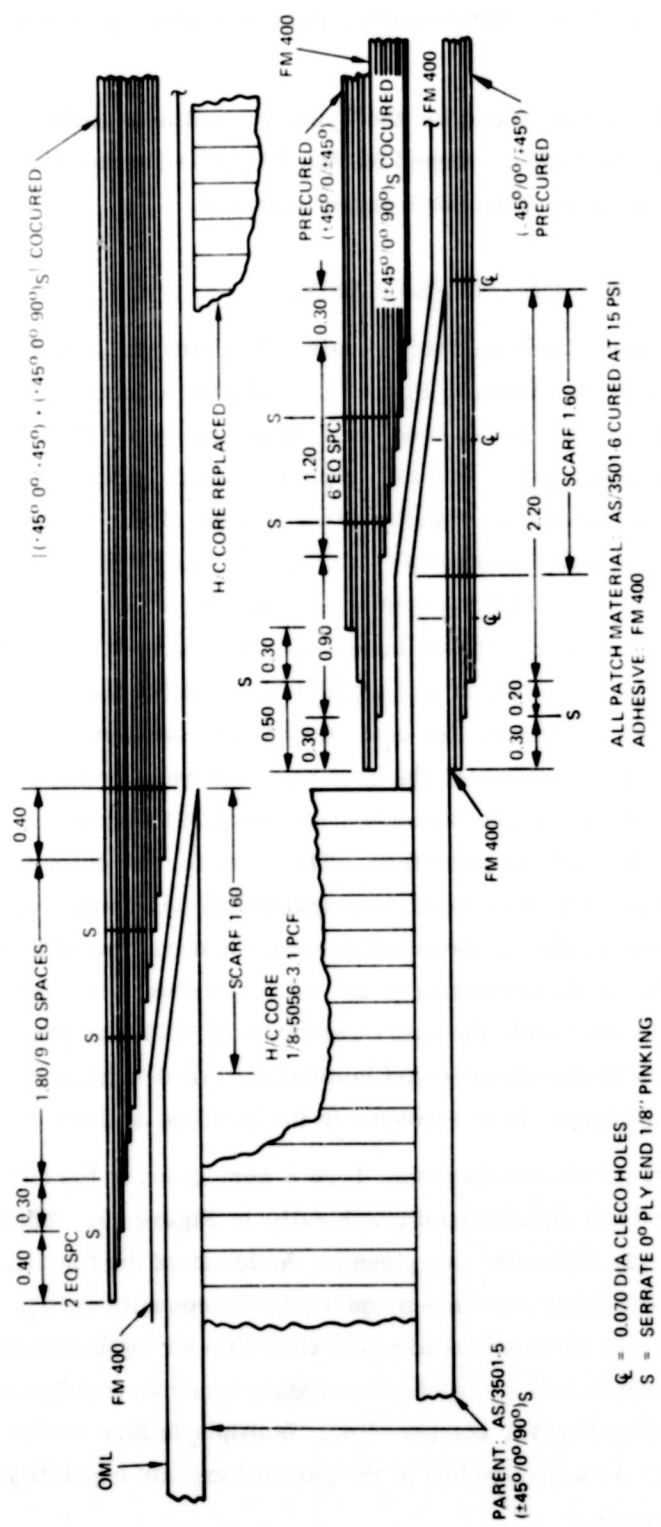
$[(\pm 45/0/90)_2]_S$ with 1 percent moisture content, the corresponding strain on loading is as shown.

The high joint efficiencies found for the Type 12 specimens indicate that the repair materials and procedures developed for the AS/3501-5 parent material also can be used effectively with the T300/5208 parent material.

3.2 SANDWICH PANEL REPAIRS (TYPES 8 AND 9)

The approach to honeycomb sandwich repair taken here was a scarf joint on both inner and outer 8-ply faces as shown in Figure 18. The inner face was repaired first following much the same procedure as used for a blind-side monolithic laminate discussed above. The blind-side patch was a five-ply symmetric precured laminate, Cleco-bonded with FM-400 adhesive to the internal surface of the inside face sheet. The open-side patch consisted of two parts: the replacement plies were cocured against the scarf surface, and the extra open-side plies were a five-ply symmetric precured laminate. The entire patch was bonded in place with FM-400 adhesive. Pressure for cocuring the inside patch was applied by installing the replacement core as a temporary plug and bagging over the core to make the vacuum seal against the outside surface. After this bond cycle, the core was removed for inspecting the inner face repair, then reinserted to make the core-foam bond. The core was then sanded flush with the inside surface of the outer face. The patch for the outer face was entirely cocured, consisting of replacement plies against the scarf and five additional plies outside the original moldline. Because no repair material could be placed against the inner surface of the outer face (i.e., all extra plies were external to the outside moldline of the outer face), the scarf was made at 0.20 inch per ply taper instead of the slope used with the single-scarf laminates. All 0-degree ply ends were serrated to alleviate the higher shear stresses in the bondline at these terminations.

The complete test results on Specimen Type 8 were given in the second interim progress report ⁽⁵⁾, and are summarized graphically in Figure 19. Allowable parent face loadings for the 8-ply laminates are given as the lower of the tension or compression allowables since both faces are loaded equally in the opposite sense. Without fatigue history, the results show essentially full allowable strength restoration in every case. Tension fatigue history is seen to slightly improve residual strength in tension and to slightly degrade it in compression. Multiple failure modes were common, indicating that the repair is well matched to the parent laminate capability where failures generally occurred.



ALL PATCH MATERIAL: AS/3501.6 CURED AT 15 PSI
 ADHESIVE: FM 400

φ = 0.070 DIA CLECO HOLES
 S = SERRATE 0° PLY END 1/8" PINKING

Figure 18. Sandwich Blind-Side Repair Face Splices Specimen Type (8)

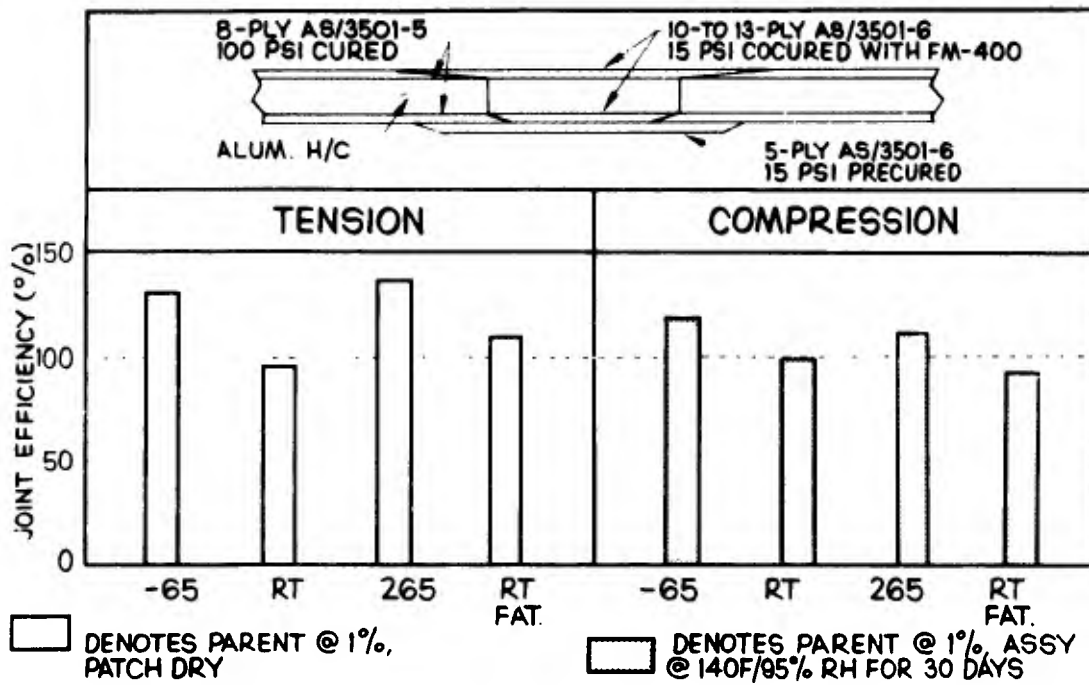


Figure 19. Blind Sandwich Joint Results

Specimen Type 9 had the same outer face repair as Type 8. The core splice at the edge of the outer parent face was the same as Type 8 except that a 1/8-5052-12.0 pcf core was used under the parent laminate. The 1/8-5056-3.1 pcf core was used under the patch, and spliced to the heavier core with FM-404 foam adhesive. The inner face repair was a simple double-lap rivet-bonded splice using two 0.020-inch-thick 6Al-4V titanium splice plates. Between the plates in the area where the inner face was removed, a potting compound, Proseal 828, was cured in place. This repair was unsuccessful for compression on the inner face because the potting compound used between the plates to stabilize them expanded upon curing, bending the splice plates apart so that they became unstable under the compressive loading. The test was stopped when this premature failure became evident on the first specimen of Type 9. To salvage the specimens for useful tests, the load was reversed, putting the outer face (with the graphite splice) in compression. Failure modes were predominantly repair compression failures of the outer face, and failures occurred at the end of the parent laminate. The ultimate load results for the Type 9 specimens tested dry were markedly lower than the results of the Type 8 specimens tested wet. Since the specimen types 8 and 9 were different in several respects, it is not possible to identify the specific reason for the low Type 9 results. In any case, the tests of

Specimen Type 9 should not be used to define the RT dry compression strength of the outer face repair of the sandwich since dry compressive strength should be at least as good as the wet compressive strength.

3.3 EXTERNAL PATCHES - (TYPES 10 AND 11)

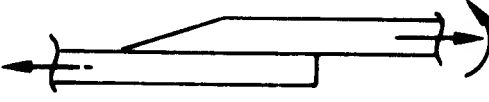
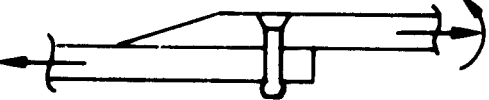

An external patch offers some advantage of simplicity where reduced strength, less aerodynamic smoothness and increased weight in a local area are acceptable. Minimum removal of the damaged material is required and scarf surface machining is eliminated.

The concept investigated used a graphite patch precured to a surface matching the outer moldline and subsequently bonded to the damaged surface. Because of the eccentricity of the load path through the patch, considerable bending occurs and additional thickness must be provided in the patch. The bending also causes a tensile stress in the adhesive at the edge of the hole where damaged material was removed. Shear stress in the adhesive also peaks at this point due to the sudden change in cross section. Several configurations were investigated empirically and the results are described in detail in the second interim progress report⁽⁵⁾. Three configurations and their test results are summarized in Table 12. Small blind rivets at 1.0-inch spacing make a significant improvement by resisting the peel force. The glass fabric and RM-400 adhesive filling the undercut make another improvement by relieving the shear stress peak. Observing the specimen as load is increased reveals no indication that the rivets resist shear until after the bond has failed.

Specimens used to develop the final configuration were tension coupons, with load applied at the center of the parent laminate on either end. The precured external patch was eccentric to the load by 0.115 inch, resulting in significant bending moment in the patch. This influenced the patch design and would cause the dimensions of the damaged panel, the size and location of the repair, the lateral support, and the load sense to affect the results. This concept is shown in greater detail in Figure 20.

Using the splice design of Figure 20, four-point-load sandwich beam specimens were produced for testing across the entire range of environmental and load conditions. The RT dry beams loaded with the repair in tension resulted in an average ultimate face loading of 6405 lb/in, while the RT dry tension coupons with the same repair design failed at an average of 5357 lb/in. This 1048 lb/in improvement was due only to the test method since the honeycomb core of the sandwich beam did not allow local bending deflections in the tension face, while deflection did occur in the coupon. A

TABLE 12. EXTERNAL PATCH CONCEPTS

FM-400 BONDED JOINTS (WIDTH = 1.00 INCH)	FAILURE MODE	JOINT EFF
 <p data-bbox="531 539 676 572">CONTROL</p>	PEEL AND SHEAR FAILURE	0.52
 <p data-bbox="428 734 778 767">ONE 1/8TH INCH RIVET</p>	SHEAR AND PATCH NET TENSION FAILURE	0.73
 <p data-bbox="394 929 813 962">ONE RIVET & GLASS FABRIC</p>	SHEAR AND RIVET HEAD PULL-THRU FAILURE	0.78

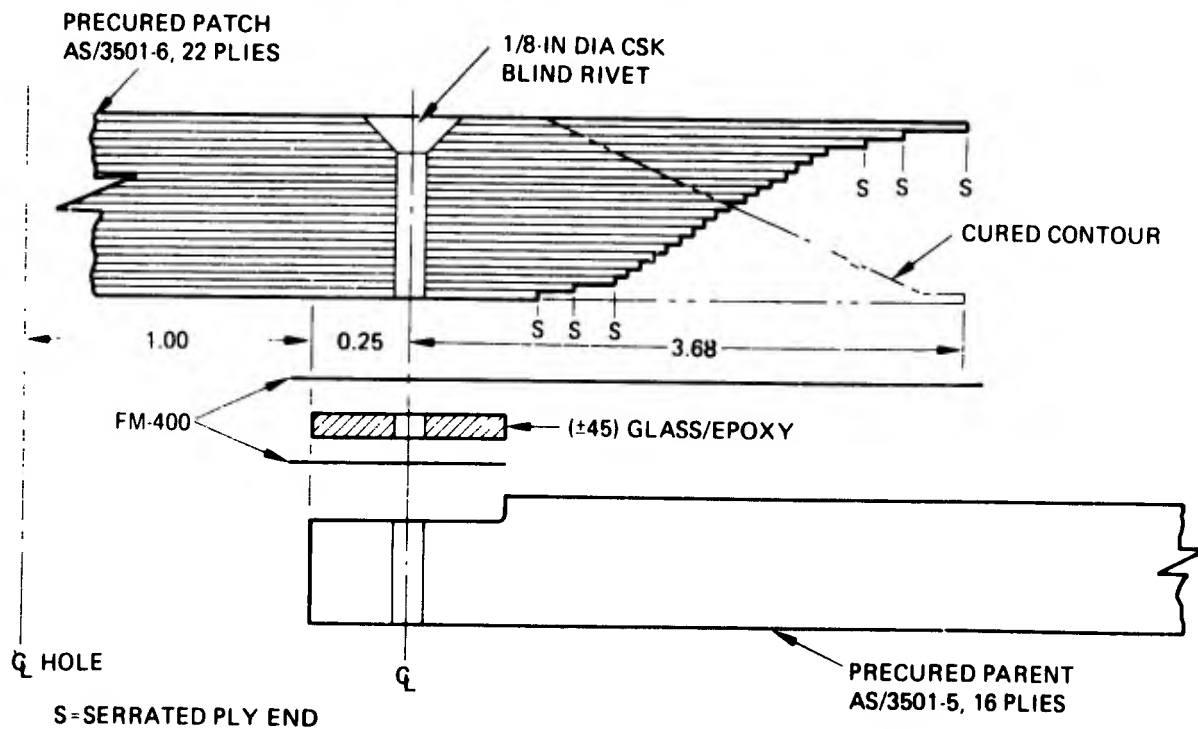


Figure 20. External Patch Repair, Type (11)

change of failure mode was also noted. The coupons failed in shear and rivet head pull-through (i. e. , at the end of the parent laminate) while the beams failed in parent laminate tension and interlaminar shear at the edge of the patch at a higher load intensity.

Test results for the external patches are summarized in Figure 21. In general, the external patch has performed quite satisfactorily in the dry condition. The 265°F test temperature modified the failure mode, but not to the detriment of joint efficiency. Tension fatigue history had little effect on either residual tension or compression strength. Failures generally appeared to have begun remote from the hole (edge of the parent laminate), which was a favorable result.

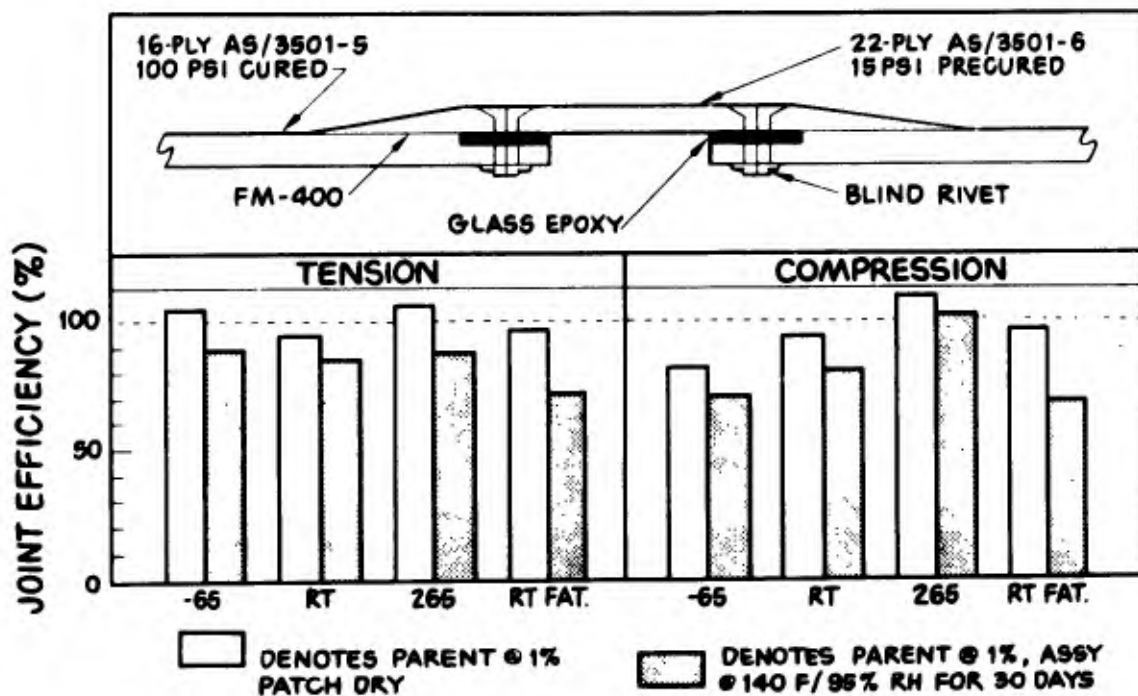


Figure 21. External Patch Joint Results

3.4 INTERMEDIATE-SIZE PANEL REPAIRS

All repair joint tests described previously in paragraphs 3.1 through 3.3 were done using coupon or sandwich beam specimens one to two inches wide in which the repair joint was one-dimensional in planform, i. e. , it extended across the full width of the specimen. To investigate the two-dimensional effects of a local repair in a wider panel, two types of repairs were applied to 4-inch diameter holes in 16-ply 12- by 48-inch $[(+45/0/90)_2]_S$ AS/3501-5 panels as shown in Figure 22. All panels

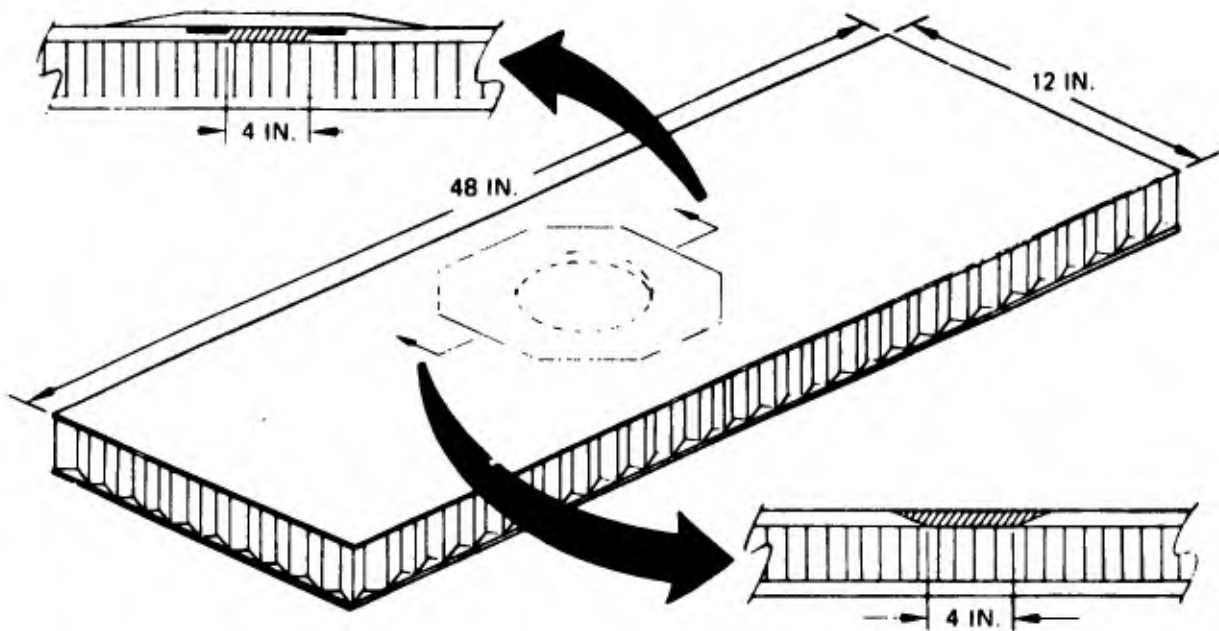


Figure 22. Intermediate Size Panel Repair Concepts

were soaked in 180°F deionized water to a 1 percent weight gain before repair work began. About 20 percent of those panels tested were conditioned in an environmental chamber at 140°F and 95 percent relative humidity for 30 days after the repair was completed. After making the repair (and moisture conditioning as required), the repaired panel and a titanium slave skin were bonded to honeycomb core completing the sandwich assembly.

The two repair configurations are direct applications of Type 1 (flush single scarf joint) and Type 11 (external patch) described previously. For the flush repair, 13 patch plies bonded to the scarfed surface were cut in a circular pattern corresponding to the shape of the cutout in the parent laminate. The six plies bonded to the outer surface and the four bonded to the inner surface were each octagons of increasing concentric size with the largest ply outside. The octagonal shape facilitated the serration cuts with pinking shears on each edge which is normal to the fiber direction of the ply. Hence, in each direction, 0-degrees, +45-degrees and -45-degrees, the two longest patch plies on each side of the laminate were serrated to reduce the adhesive shear at the ends of these plies for tension, compression and panel shear loading.

Two methods of machining the sloping scarf joint surface were compared. One panel was cut on a milling machine with precise control of the tolerances and smooth-

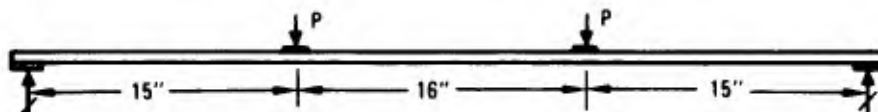
ness. Three other panels were cut using hand routers and sanding tools which resulted in much less precise control. Subsequent test results showed no effect of the machining techniques used.

Static and fatigue tests on ten external patch repair specimens and four flush scarf joint repair specimens were conducted as four-point beams as indicated in Table 13.

TABLE 13. INTERMEDIATE SIZE PANEL TEST PROGRAM

COMPONENT APPLICATION	PARENT LAMINATE	DAMAGE/PATCH CONFIGURATION	DRAWING NO.	QUANTITY PER TEST CONDITION			
				RTD STATIC		RTW STATIC COMP.	RTD FATIGUE COMP.*
				TENSION	COMPRESSION		
GENERIC SANDWICH	16-PLY G/E [(1:45/0/90) ₂] _s	THROUGH-PUNCTURE CIRCULAR HOLE					
		● FLUSH PATCH	SRO13021	3	1	-	-
		● EXTERNAL PATCH	SRO13024	3	3	3	1

*MAXIMUM PANEL FACE LOADING = -2820 LB/IN. AT A STRAIN OF -4570 μIN./IN. DURING 2 LIFETIMES OF F-5E WING HISTORY.



Detailed descriptions of the specimen fabrication and testing are given in the third interim progress report⁽⁶⁾, and are summarized here. For the external patch repair specimens, the tension loaded panels sustained 81 percent of the parent allowable, failing the repair near the patch edge. No incipient failure was noted except for the usual faint sounds near ultimate load. The somewhat poorer performance of the panels as compared to the one-inch wide beams of the same patch cross-section is explained in part by the strain gage data. Gages near the end of the patch read 20 percent higher than a gage remote from the patch and near the edge of the panel. Rather than uniformly distributing load around the hole, the effect of the patch was to create a "hard spot", drawing load toward the repair and resulting in a mild stress concentration. In addition to the replacement plies, the patch design included six extra zero-degree plies to deal with the local bending due to patch eccentricity. The termination of the second and third longest zero-degree plies, though they are stepped generously (0.30-inch/ply) and serrated, imposes extra load on the longest patch ply since it is in

intimate contact with the terminating plies. Inwardly, the nearest zero-degree ply in the parent laminate is on the opposite side of the adhesive and below the ± 45 -degree plies. The tension failure suggests that failure began where the longest ply joggles over the end of second longest ply. The failure may begin at the joggle by breaking the surface patch ply which then causes the patch to peel, taking parent material with it, finally failing the weakened section through the hole.

The external patch compression panels, both dry and wet, did not perform as well as the tension panels nor as well as the one-inch wide beams. The discussion of stress distribution given above applies here as well as the explanation of this relatively poor performance. In addition, the mode of failure of the compression panels was rather surprising. Compression failures began consistently at about 25 percent of the parent allowable load. The surface zero-degree ply popped loose locally where it joggled over the end of the next longest ply. While the joggle is only a 0.0055-inch step, it nevertheless appears to have a significant effect on the ability of the 0-degree ply to resist local compressive failure. Once a local failure has occurred, it grows with increasing load and is accompanied by progressive failures until the entire patch is peeled off.

Moisture conditioning of the external repair had no deteriorating effect on the results. This would follow as the failure development pattern appears to be the same as wet and dry repairs.

Concern over fatigue loading would appear justified where the compression loads would be high enough to cause incipient failure such as experienced on the static tests of the external repairs. For this reason, the fatigue loads were applied in compression using the same F-5E fatigue spectrum used previously in tension with a peak compression load strain of $4570 \mu\text{in}/\text{in}$. The outer ply did not pop loose during fatigue loading as might have been expected, although the peak fatigue load was 1.40 times the load at which this incipient failure occurred during static compression loading. In the residual compression strength test after fatigue loading, incipient failure began only after the load had reached 45 percent of the parent compression allowable. This improvement in the onset of incipient failure corresponds to a like improvement in ultimate failure load: 84 percent efficiency as compared to 66 percent for the dry repairs with no prior fatigue history.

Four panels were repaired with a nearly flush patch on a scarfed hole. The tension loaded panels performed very well at 109 percent joint efficiency. All failures were in parent laminate tension beyond the end of the patch. Only the usual faintly

audible "pop" sounds were heard near ultimate load and no visible incipient failure was observed prior to failure.

The one flush scarf repair tested in compression achieved 92 percent of the parent compression allowable. Failure occurred suddenly with no audible or visual signs of incipient failure or early deterioration of the repair. The failure tore the patch away before finally crushing a section through the 4-inch diameter hole.

The test results for the intermediate size panels are summarized in Figure 23.

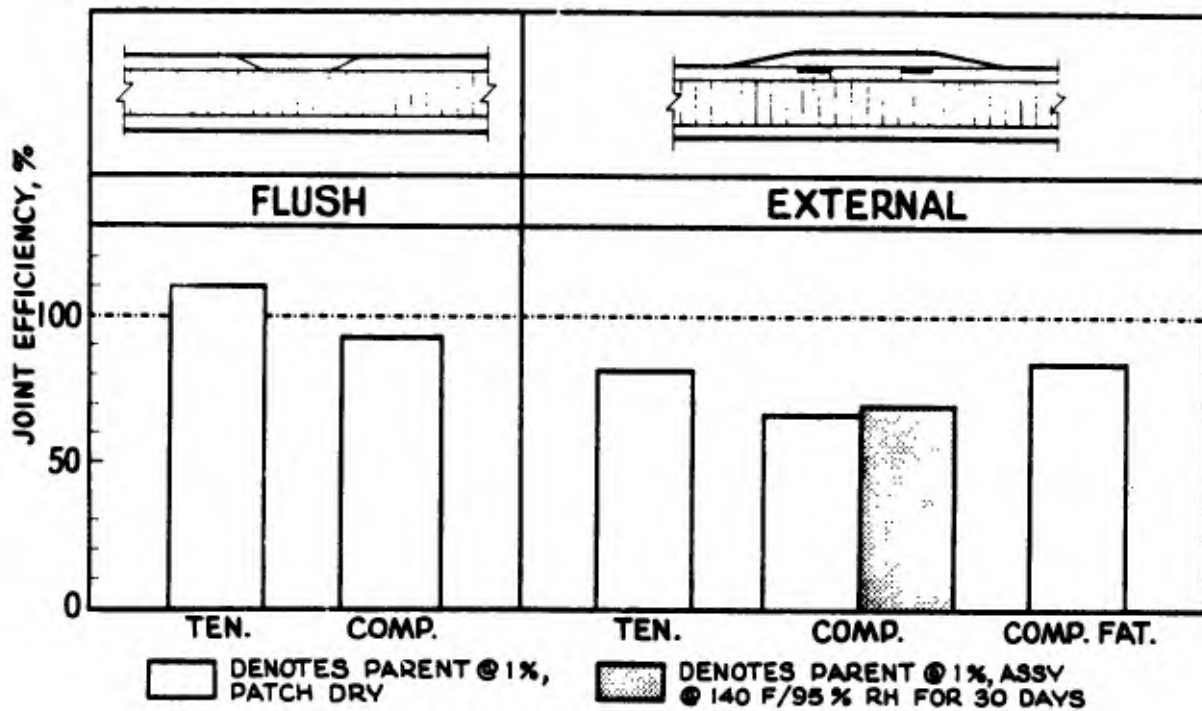


Figure 23. Intermediate Size Panel Results

SECTION IV

LARGE SCALE DEMONSTRATION COMPONENTS

The objective of this portion of the program was to apply the repair techniques developed earlier to five large-scale demonstration components. Each of these components was a 19 x 60-inch panel and each involved a different repair problem. Each panel was tested in a box beam imposing uniform bending over the central five-foot section and torsion in some cases as shown in Figure 24. All five panels were completely dry (i. e., no moisture conditioning), all were tested at room temperature and all were failed with uniform bending only on the test section, resulting in tension on the panel.

The question of what the ultimate allowable load of mechanically fastened panels should be requires some attention. For the small scale coupons and beams and for the intermediate size panels, the failure loads have been compared to the parent

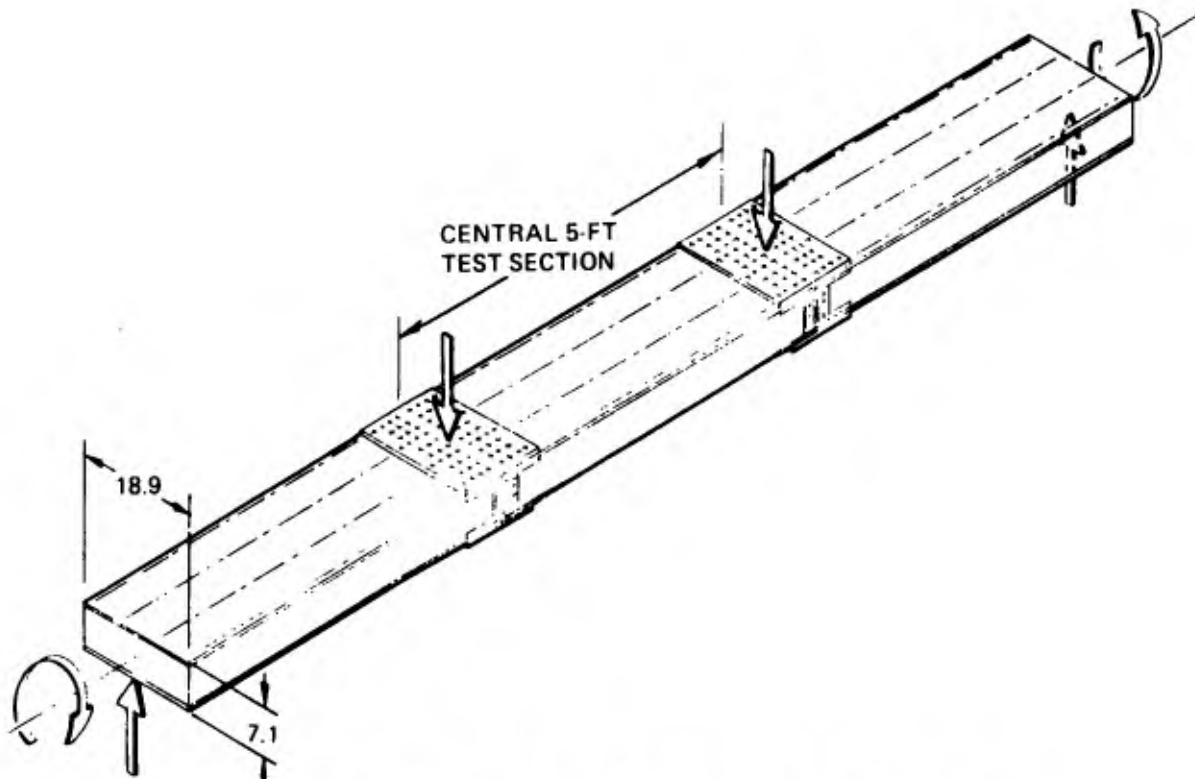


Figure 24. Large Scale Demonstration Component

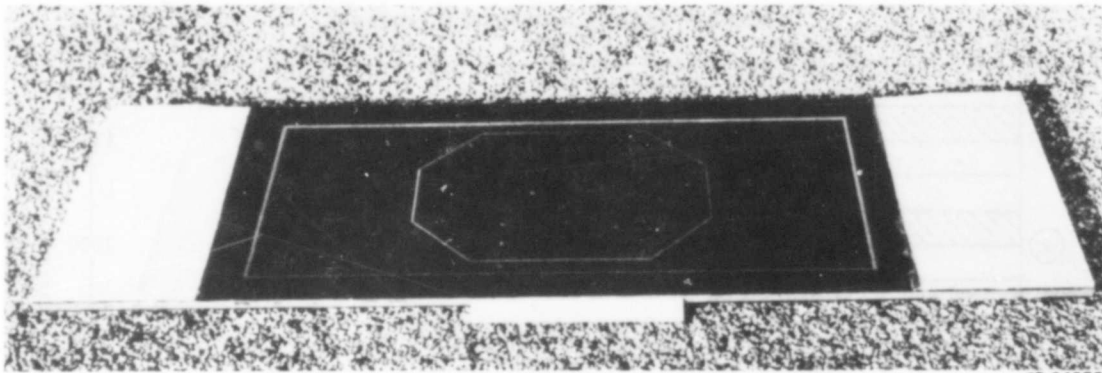
laminates ultimate allowable. (For specimen Type 4 typical data was used since only a small quantity of data was available.) Since all of the demonstration components are mechanically fastened to the spars of the box beam with 3/16-inch fasteners, the parent laminate allowable can never be realized in the panels. The work of Verette and Labor⁽⁷⁾ shows that unloaded 3/16-inch 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 -degree plies will result in a higher relative strength. Therefore, the panel design ultimate allowables (D. U. A.) have been defined as 0.5 times the parent material ultimate tension allowables, except in one case where the D. U. A. was obtained more directly related to the simulated component.

4.1 PANEL ONE: HONEYCOMB SANDWICH, ACCESSIBLE BOTH SIDES

The faces of Panels One and Two were each 8-ply laminates $(\pm 45/0/90)_S$ with 0.50 inch thick 5056 aluminum honeycomb core, 1/8-inch cell size, 8.1 pcf. The panel room temperature design ultimate allowables are: tension 1715 lb/in. per face, compression 1900 lb/in. per face and shear 1418 lb/in. per face based on the data in Table 1. The tension and compression allowables were derived using the factor of 0.5 for the effect of holes but the shear allowable does not contain this factor. This is because the panel edge was built up with a 16-ply doubler on each face. This material does not significantly effect the longitudinal strains but has a direct effect on shear strains since the shear flow is constant whether within the built-up area or outside of it.

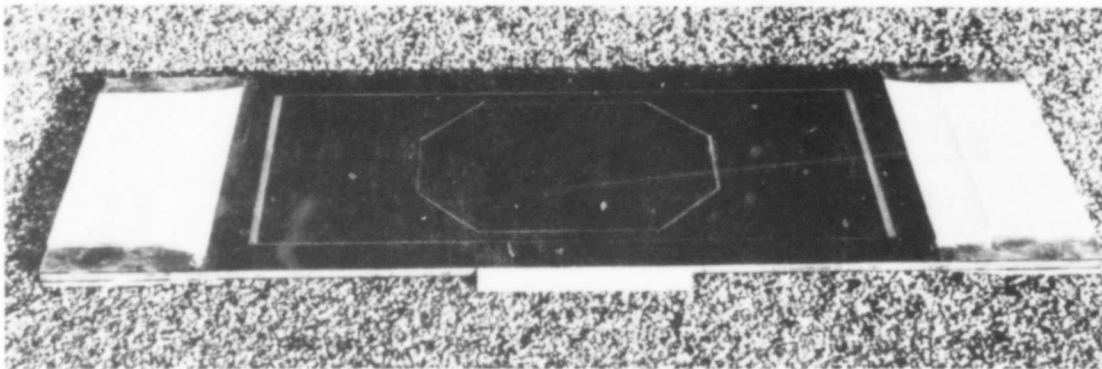
The repair and testing of Panel One is described in detail in reference (6) and is summarized here. An oval hole 6.6 x 12.0 inches completely through the panel was repaired with access to both sides. The core was replaced and spliced. Then both faces were repaired with a bonded and cocured patch which 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 panel faces were scarfed adjacent to the hole on 36/1 slope using a portable drum sander. A track was provided to guide the tool and the scarf angle was fixed by the track and tool construction. The sanding drum was shrouded and provided with a vacuum source to remove the dust. The completed repair is shown in Figure 25.

The panel was initially loaded in tension by applying bending to the box beam. Loading was terminated when the maximum measured strain reached 7333 μ in./in. (2/3 of the parent laminate ultimate tension strain allowable in Table 1). Two additional loading conditions were subsequently applied, i. e., compression caused by



a) EXTERNAL SURFACE

78-04086



b) INTERNAL SURFACE

78-04081

Figure 25. Repaired Panel One

beam bending and shear produced by beam torsion loading. For each, loading was terminated if either (1) the maximum measured strain reached $7333 \mu\text{in./in.}$ (for compression) or (2) the load reached $4/3$ of the panel design ultimate allowable (as it did for shear loading). The recorded load cell data later indicated the shear load to be 131 percent N_{xy}^{su} . The panel showed no indication of failure at the completion of the three preliminary test conditions.

Bending was again applied to the box beam in load condition four, until tension failure of the panel. Failure loading in the 8-ply outer face was 2390 pounds per inch. The maximum axial strain at failure was $9670 \mu\text{in./in.}$ The results of all load conditions are shown in Figure 26 in terms of percent of allowable load and strain. The nonuniform strain distribution, shown in Figure 27, became an advantage in this particular case since it reduced strain along the panel edge where the attachment holes are located. The failure, shown in Figure 28, occurred remote from the patch.

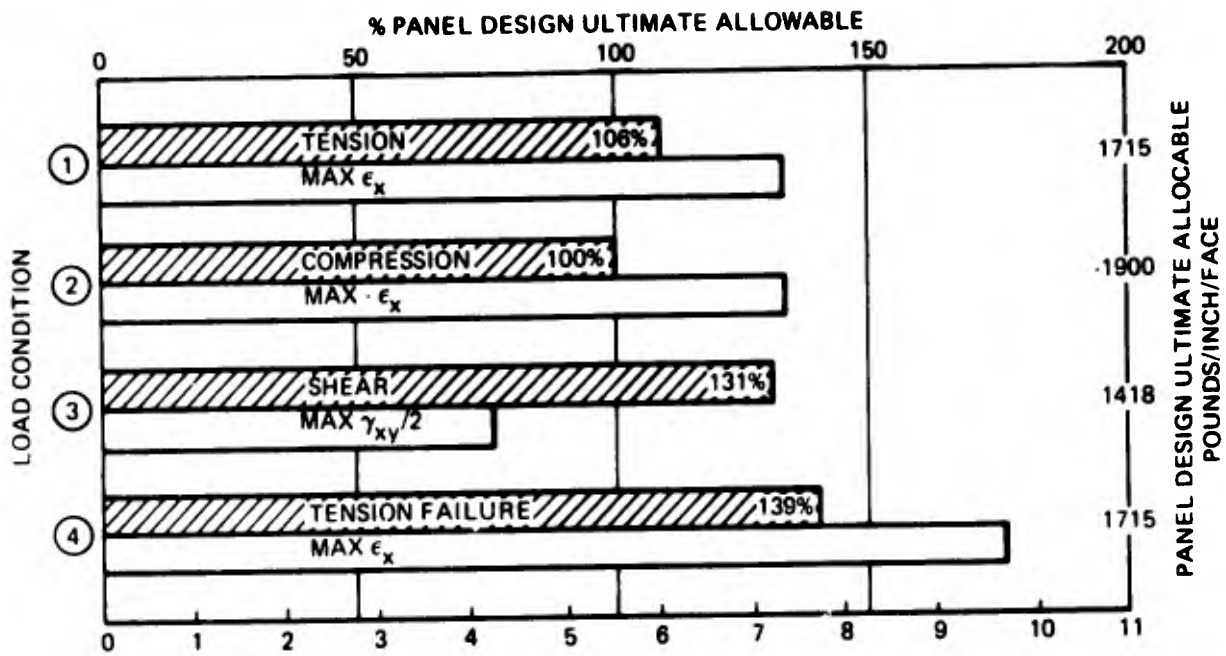


Figure 26. Panel One Test Results

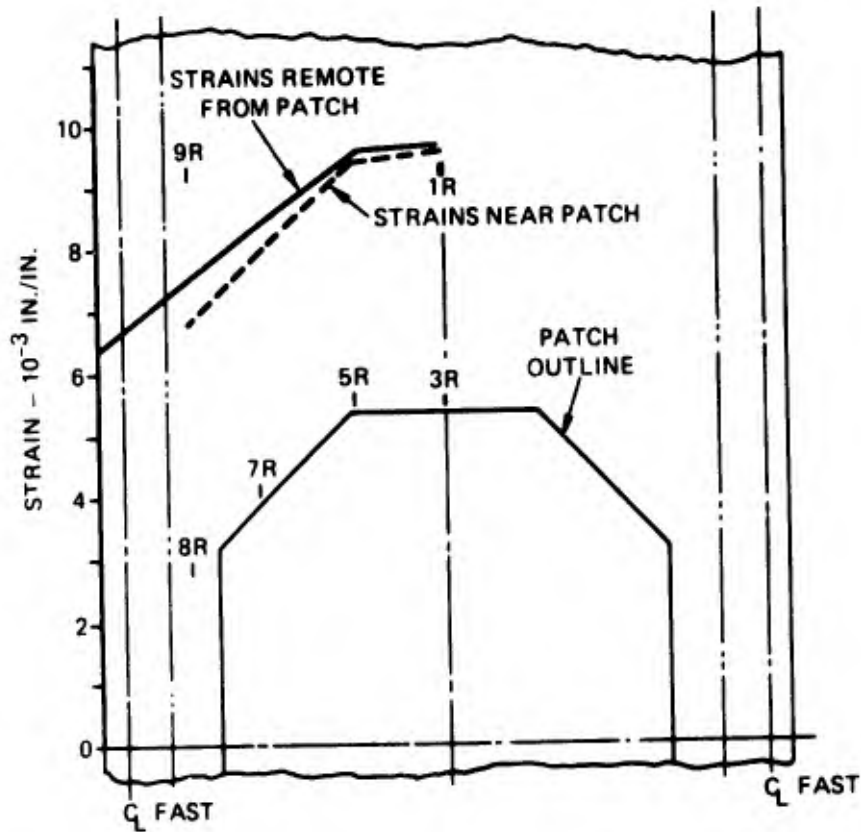
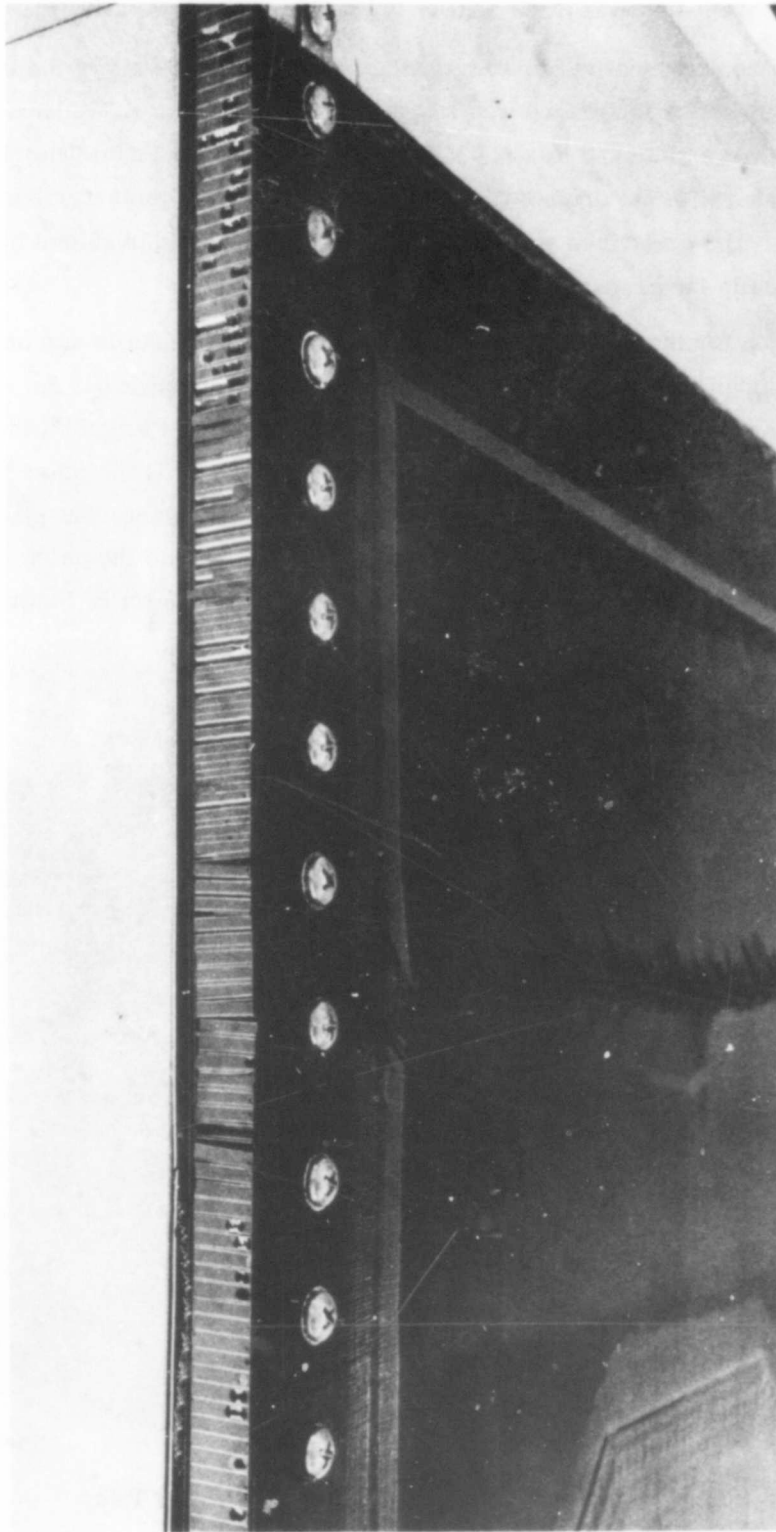


Figure 27. Panel One Failure Strain Distribution



78-01623

Figure 28. Panel One Tension Failure at 139% of Design Ultimate Allowable

4.2 PANEL TWO: HONEYCOMB SANDWICH ACCESSIBLE ONE SIDE ONLY

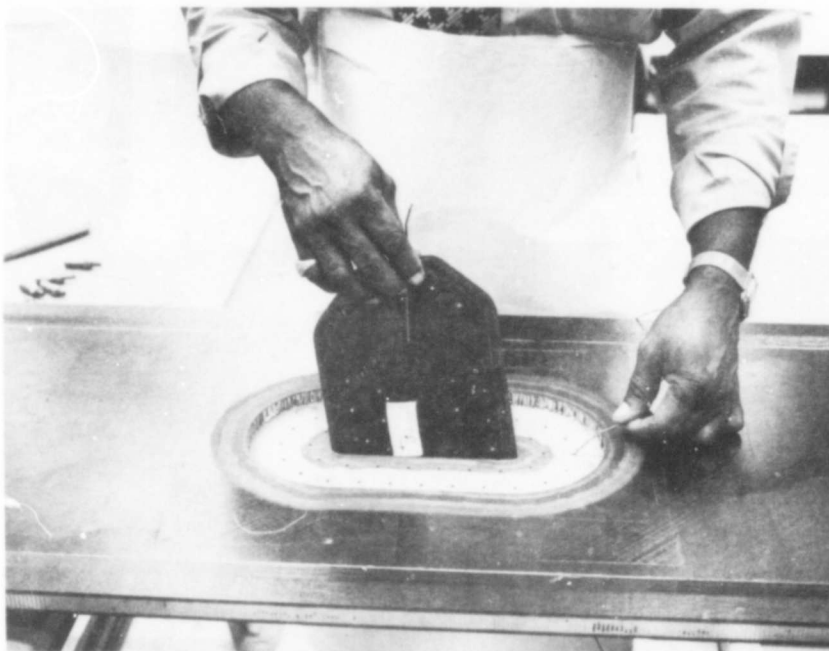
Panel Two was identical to Panel One except for the repair of the inner face. The penetration of the inner face was a longitudinal hole 1.0 x 6.4 inches. The outer face and core were removed to a 6.6 x 12.0 oval hole (as on Panel One). The inner face was scarfed with the drum sander guided by the hole in the outer face, as shown in Figure 29. The outer face was scarfed with the drum sander guided by the 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. Making and assembling these details may seem complex but the results have been rewarding. 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-inch diameter 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 as shown in Figure 30 (adhesive omitted for clarity).



77-05999-8

Figure 29. Scarfing the Inner Face of Panel Two



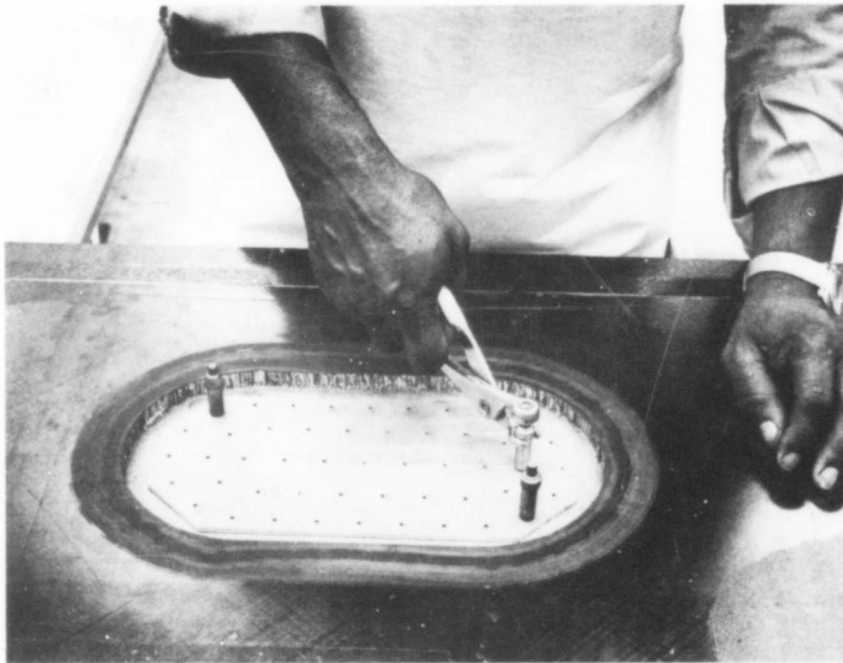
77-01296-10

Figure 30. Installation of Drilled Precured Blind Side Doubler

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 Figure 31, to provide pressure during the bonding of the blind side patch. The clecos had been coated with Freecoat 33 and baked at 350°F for one hour. 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 is

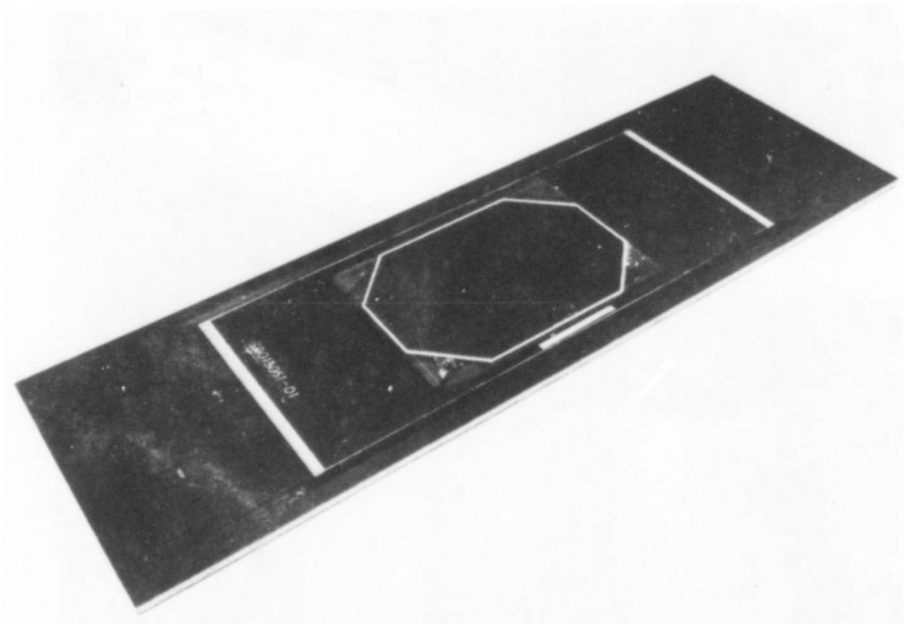


78-01297-4

Figure 31. Installation of Clecos for Bonding of Blind Side Doubler

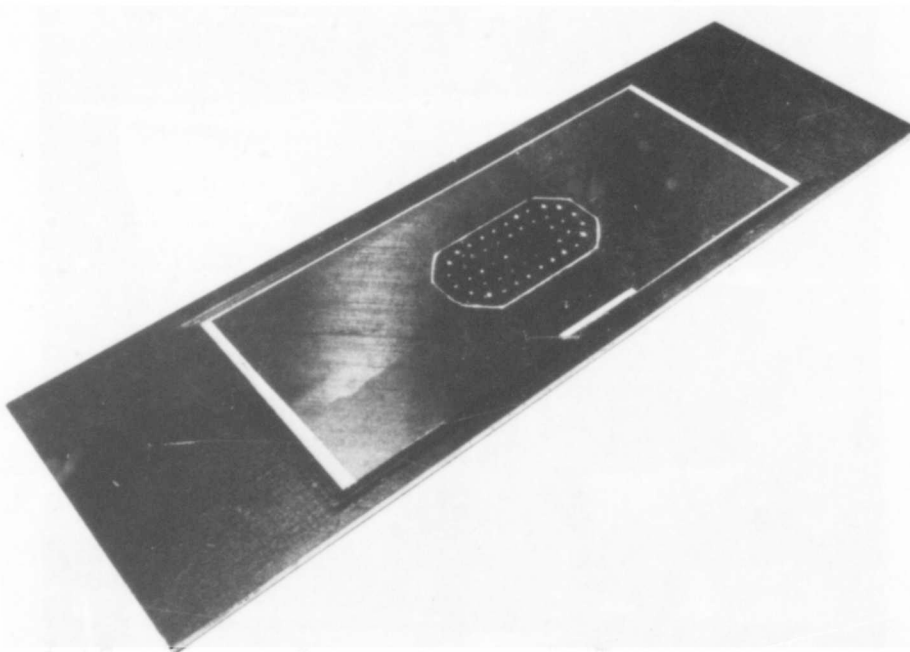
seen in Figure 32. The edges of the blind side patch detail were imperfectly bonded at best. Inspection of this patch from the inside is a luxury not available to an airplane repairman. The repair was therefore tested as-is with no attempt to improve it.

Panel Two was loaded to failure in tension resulting from positive bending on the box beam. The panel design ultimate allowable was defined in the same way as for Panel One, i. e. 1715 pounds per inch per face. Panel Two failed at 2095 pounds per inch in the outer face or 122 percent of design ultimate allowable. Maximum strain at failure was $8925 \mu\text{in./in.}$ 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. Photos of the failure, viewed from the outside and inside are shown in Figure 33. 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 which is capable of high strength restoration a considerable margin is available to allow for reduced quality standards.



a) OUTER SURFACE

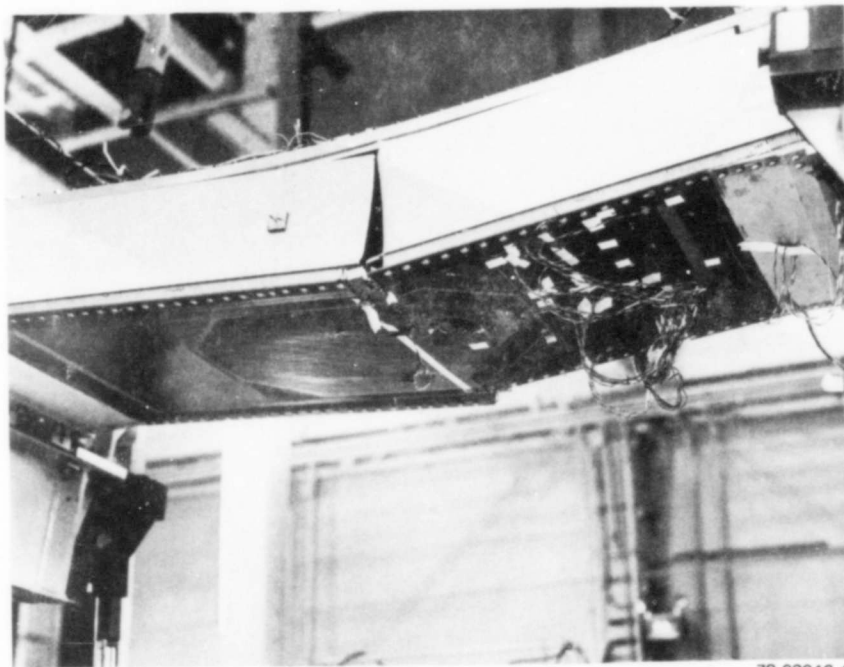
78-02221-7



b) INNER SURFACE

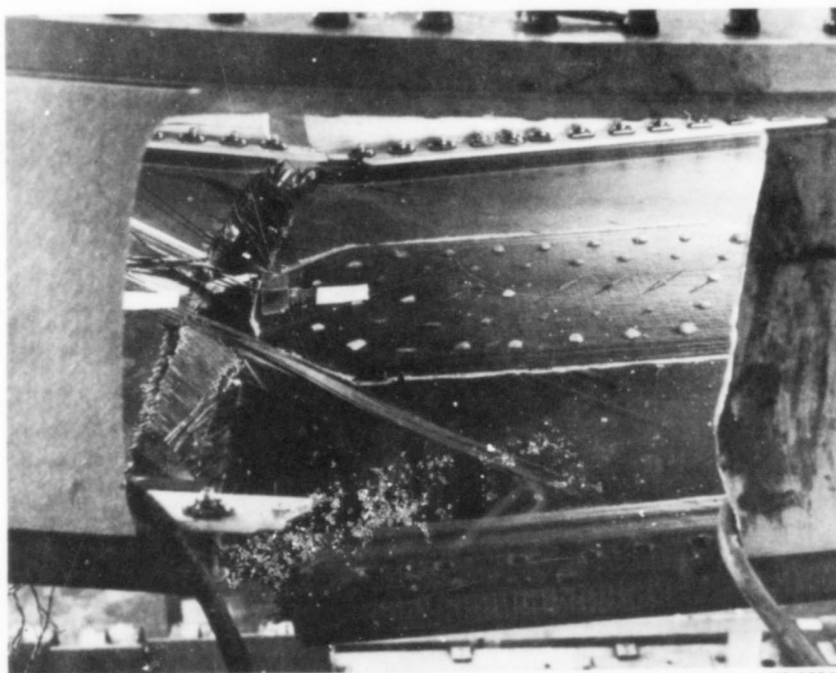
78-02221-8

Figure 32. Panel Two Completed Blind Side Sandwich Repair



a) EXTERNAL VIEW

78-03016-4



b) INTERNAL VIEW

78-02977

Figure 33. Panel Two Failure at 122% of Design Ultimate Allowable

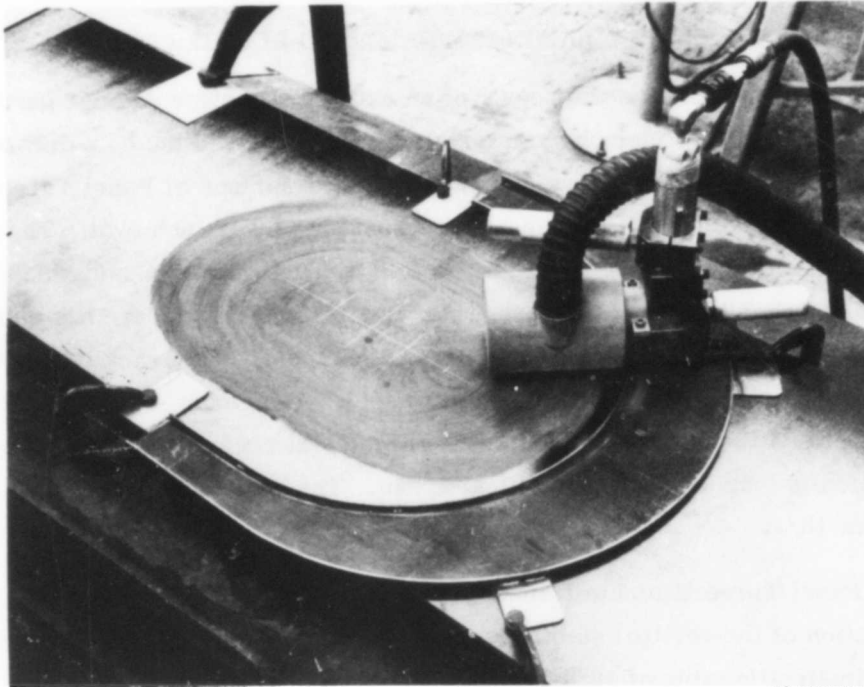
4.3 PANEL THREE: 50-PLY LAMINATE DAMAGED 21 PLYS DEEP

Panel Three 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 Three to a maximum depth of 0.120 inch (21 plies) throughout an 8 by 12 inch oval. The edge of the cutout area was then scarfed to a larger 12.2 by 20.4 inch oval using the drum sander with the variable angle track shown in Figure 34. 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.

The Panel Three laminate $[\pm 45/90_2/\mp 45/\pm 45/0_4/\mp 45/\pm 45/0_3/\mp 45/\pm 45/0_2]_8$, duplicates a section of the vertical stabilizer skin of the B-1 airplane. A parent laminate tension ultimate allowable of 18,270 pounds per inch at room temperature was determined from the results of six tests. (12) The skin is obviously stiffness designed, considering the maximum limit loads shown in Table 14.

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 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 zero, +45 and -45 degree directions. The order was slightly changed from the parent only so that the zero-degree 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 Figure 35.

Testing of Panel Three proceeded first by applying the limit loads of Table 14 (neglecting N_y) with no indication of failure. Load condition four was positive bending to failure at an average tension panel loading of $N_x = 14,200$ pounds per inch. The maximum axial strain recorded in the panel and projected to failure was $7720 \mu\text{in./in.}$ Failure occurred remote from the repair, as shown in Figure 36. Again using the 50 percent strength reduction factor for the skin/spar attachment holes the panel ultimate allowable would be 9135 pounds per inch. Hence, the repair sustained 155 percent of the panel design ultimate allowable strength.

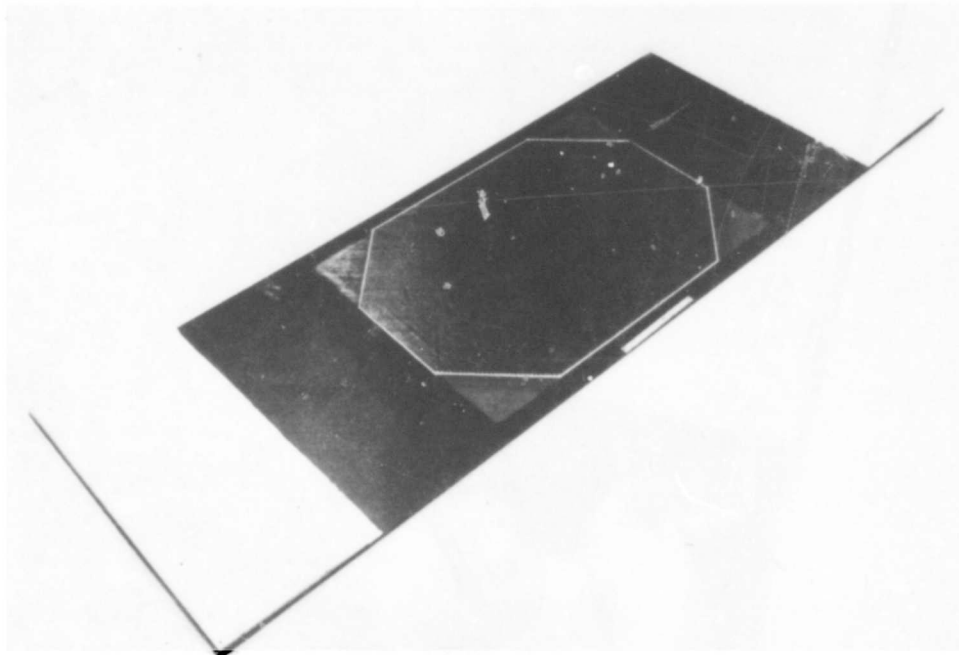


78-00507-5

Figure 34. Scarfing the Damaged Cutout on Panel Three

TABLE 14. LIMIT LOADS FOR SIMULATED VERTICAL STABILIZER SKIN

LIMIT LOAD CONDITION	N_x LB/IN	N_y LB/IN	N_{xy} LB/IN
(1) MAX. POS BENDING (FEM-1)	3503.	83.	546.
(2) MAX. NEG BENDING (FEM-1)	-3503.	-83.	546.
(3) MAX. TORSION (FEM-2)	-2380.	49.	1027



78-02221-9

Figure 35. Panel Three Completed Partial Thickness Repair

4.4 PANEL FOUR: 50-PLY LAMINATE ACCESSIBLE BOTH SIDES

Panel Four represents the repair of a complete penetration damage of a highly loaded laminate. The laminate was identical to Panel Three except for the repair. A four-inch diameter 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, seven were at zero-degree orientation and four were at ± 45 degrees, all interspersed among the 50 replacement plies. In the oven cure the panel was mistakenly supported on several steel bars rather than on a continuous plate. One bar crossed the center of the repair so that the patch sagged over the bar and became cured in this form. The patch inner surface had sagged approximately 0.080 inch in a length of 0.4 inch. The depth of the fiber waviness was not known since the outer surface was smooth.

The testing of Panel Four proceeded in the same manner as for Panel Three. No indication of failure was seen from the three limit load conditions of Table 14. Positive bending to failure resulted in an average panel loading of $N_x = 14,200$ pounds per inch, almost identical to the load required to fail Panel Three. The maximum

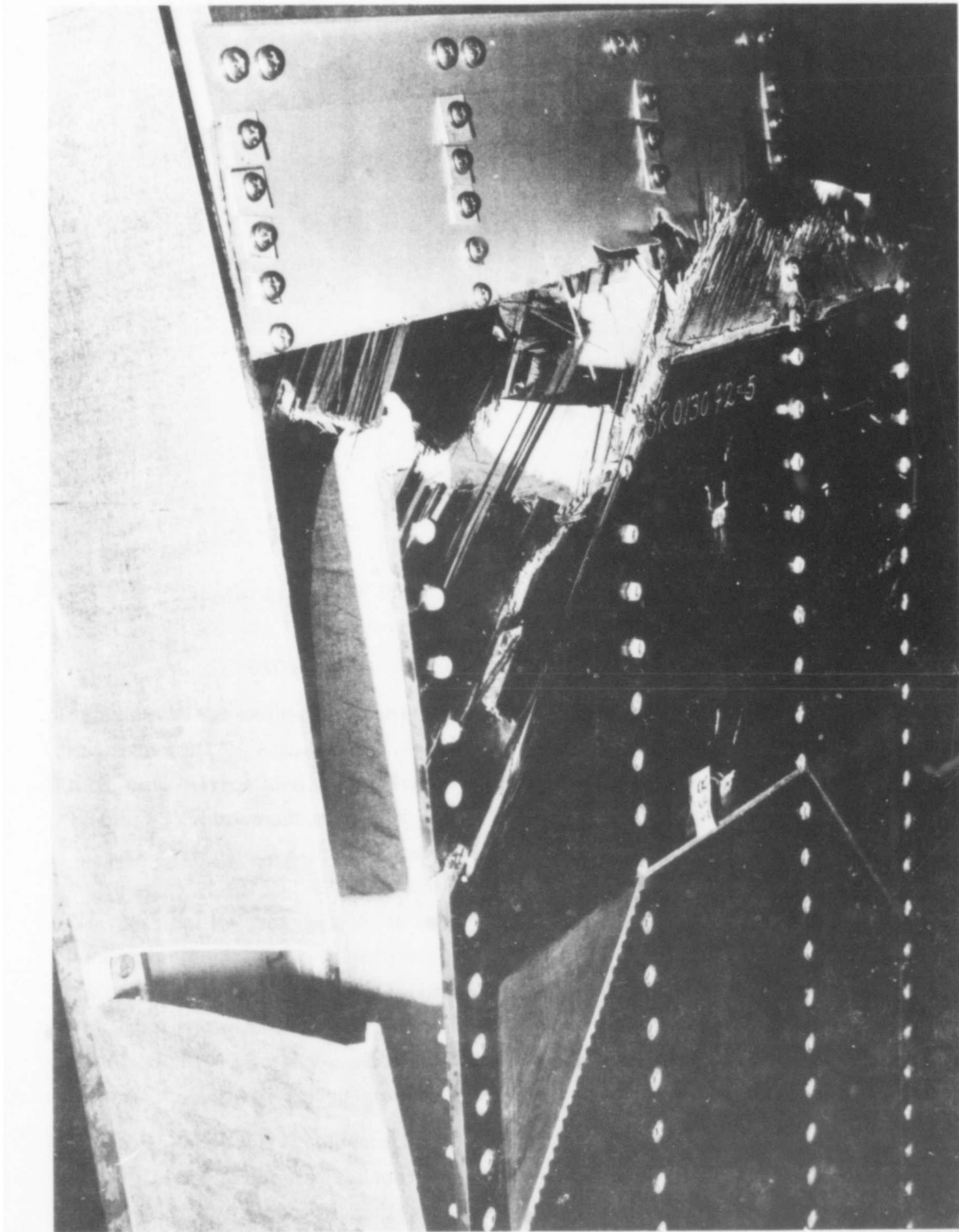


Figure 36. Panel Three Failure at 155% of Design Ultimate Allowable 78-04464-10

strain at failure was $6920 \mu \text{in.}/\text{in.}$ with a more uniform distribution than in Panel Three. Failure occurred through the repair but not involving the area of waviness, as shown in Figure 37. In spite of the irregularity in this repair the results were quite satisfactory.

4.5 PANEL FIVE: 64-PLY BORON-GRAPHITE/EPOXY HYBRID LAMINATE

Panel Five was an exact representation of a section of the horizontal stabilizer skin of the B-1 airplane with local build up over spar caps using unidirectional boron/epoxy strips. The typical cross section is seen in section B-B on sheet 2 of drawing SR-013041 presented as Figure 38. The assumed damaged extended 24 plies deep as shown in the transverse section F-F on sheet 3. This cutout removed 11 of the 28 plies of boron over the two interior spar caps plus all the associated graphite between the spar caps as shown in section F-F. The scarf in this section involved two slopes to minimize cutting of shear plies in the panel. Ideally, only the 13 plies were to be cut by the transverse scarf on a slope of 0.10 inch per ply. The longitudinal section through the damage removal area is shown in section E-E on sheet 3. The scarf length is 7.20 inches or 0.30 inch per ply. The panel after material removal is shown in Figure 39.

The embedded fibers in this panel complicated the material removal significantly. A diamond-coated end grinder was used to make the 9.44-inch diameter 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 joint 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 inch.

A decision was made to modify the patch ply sizes slightly from that shown on the repair drawing to accommodate these variations. The presence of the boron strips makes the ply overlaps of the scarf joint more critical for this panel than for previous panels with constant ply orientations across their full width.

By placing a semi-transparent 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

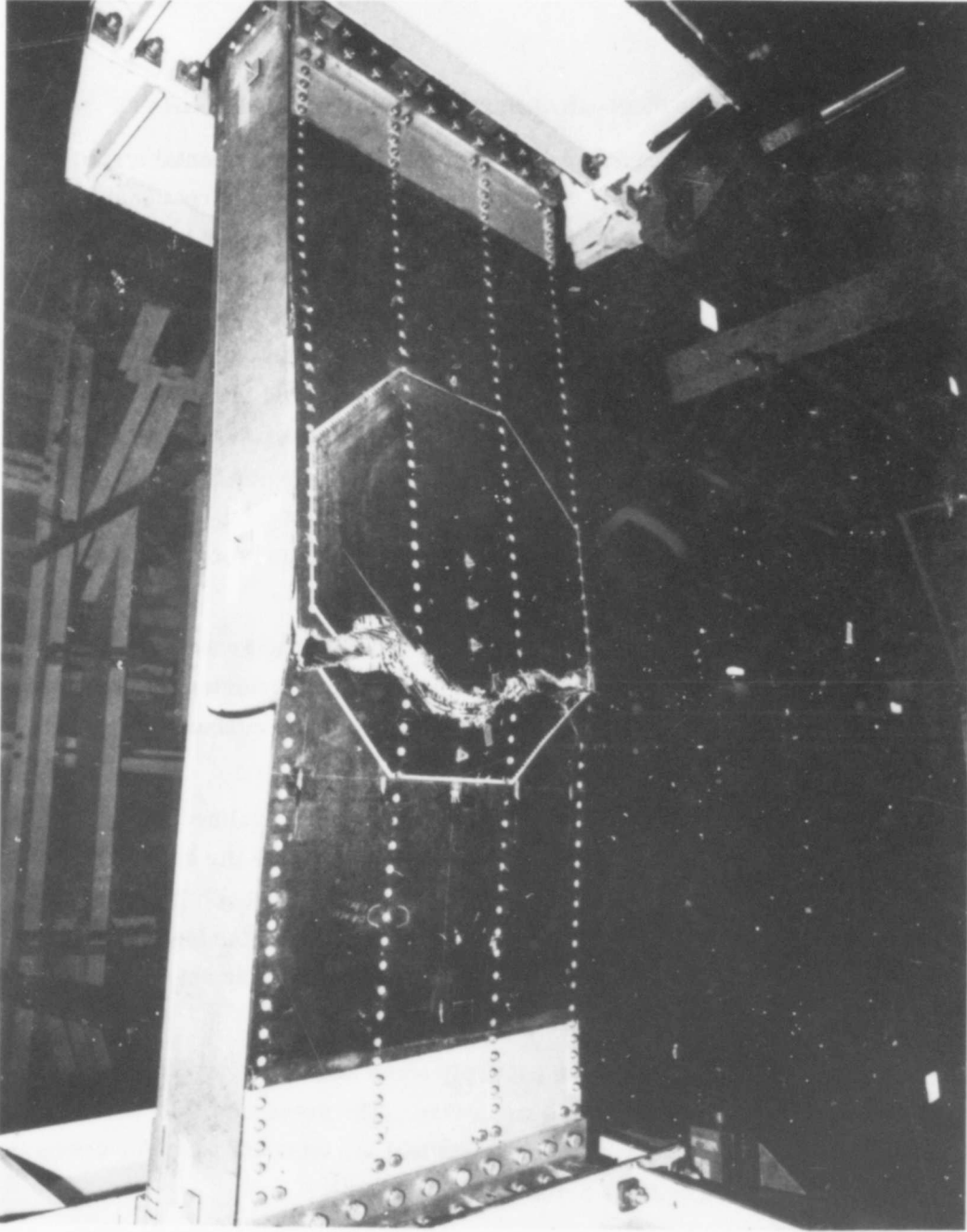


Figure 37. Panel Four Failure at 155% of Design Ultimate Allowable

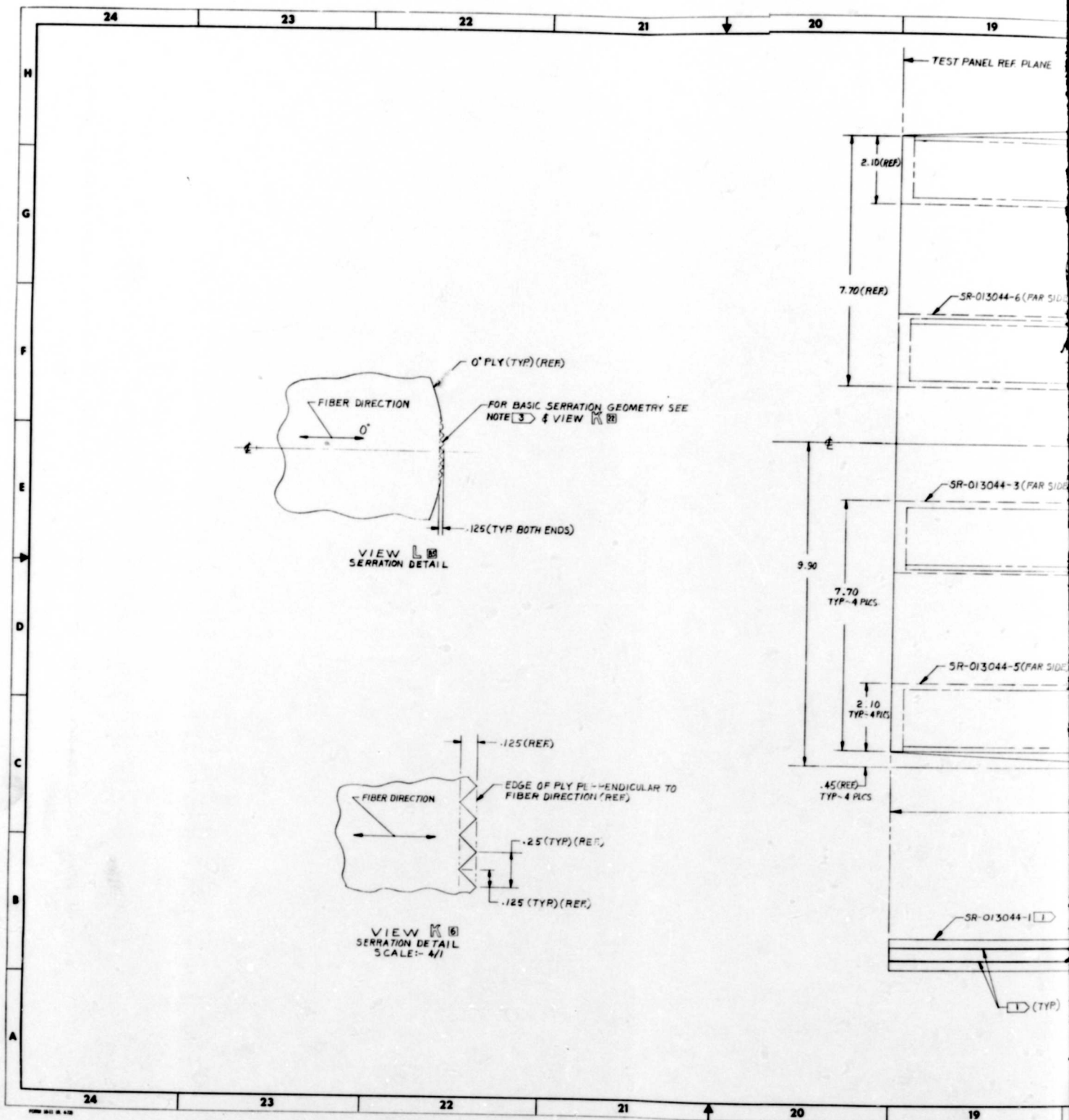


Figure 38. Panel Five Repair Drawing (Sheet 1 of 4)

SYN

SR-013044-1 (NEAR SIDE) (REF)

11.92

4.99R

4.72R

6.29R

CONTOUR 'C' (REF)

CONTOUR 'B' (REF)

CONTOUR 'A' (REF)

30.00

-01 REPAIRED PANEL ASSY (5)
VIEW LOOKING AT OUTER SURFACE

LAMINATE - SEE DETAIL

SR-013041

2

12

11

10

9

8

7

19.00 (19 EQ SPKS)
TYP - 4 PLCS

Ø4X.098) PILDOT HOLE FOR 3/8 DIA JO BOLT
(TYP) 156 PLCS. TO MATCH SR-013043

TEST PANEL REF PLANE

11.00 (REF)

SR-013044-1 (NEAR SIDE) [1]

1.01

11 (REF)

6.61

SR-013044-5 (FAR SIDE)

SR-013044-3 (FAR SIDE)

SR-013044-6 (FAR SIDE)

6.61

SPAR - SR-013043-1 (REF) - TYP 4 PLCS.

1.01

- [8] CURE -03 LAMINATE PER INSTRUCTIONS GIVEN IN NOTE 13 OF DA
 - [7] CURE -05-07-09&-11 AS FOLLOWS:- APPLY 15PSI VACUUM BAG PRESSURE HOLD FOR 2 HOURS AND COOL TO 150°F BEFORE REMOVING VACUUM
 - [6] -05,-07,-09&-11 TO BE CURED DURING THEIR BONDING OPERATIONS
 - [5] USE PORTABLE TOOLS FOR MACHINING -03 LAMINATE. MACHINING #1
 - [4] IN AREA INDICATED POT FASTENER HOLES IN LAMINATE WITH A
 - [3] SERRATE INDICATED EDGE OF PLY WITH PINKING SHEARS (W/SH)
 - [2] PANEL LONGITUDINAL EDGES INDICATED TO BE PARALLEL WITH
 - [1] COLD BOND TABS WITH EP9320(NA)-1382 PER PROCESS SPEC
- NOTES:-

SR-013041

12

11

10

9

8

7

3

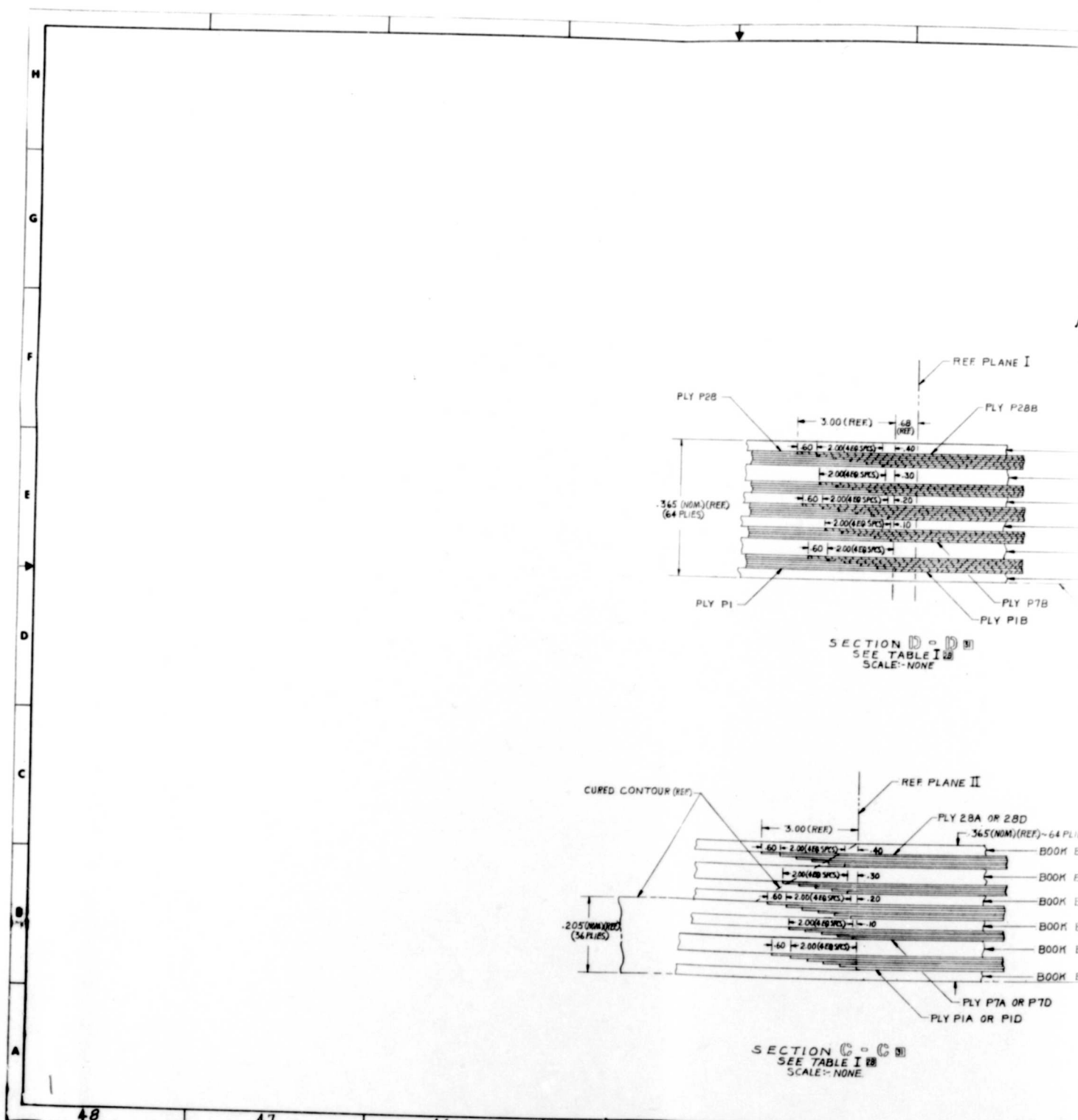


Figure 38. Panel Five Repair Drawing (Sheet 2 of 4)

• MICROFILM OVERLAP AREA •

TEST PANEL REF PLANE

REF PLANE I

REF PLANE II

1.01

BOLTS (REF)

6.61

BOLTS (REF)

19.80

BOOK B6 (REF)

BOOK B5 (REF)

BOOK B4 (REF)

BOOK B3 (REF)

BOOK B2 (REF)

BOOK B1 (REF)

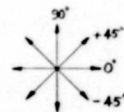
TOOL REF PLANE

9.90

BOLTS (REF)

PLIES PIC-P28C

6.61



BOLTS (REF)

PLIES PIC-P28C

PLIES EI-ER8

1.01

.45
TYP- 4 PLCS.

ES

B6 (REF)

B5 (REF)

B4 (REF)

B3 (REF)

B2 (REF)

B1 (REF)

TOOL REF PLANE

2

SR-013041 2

42

41

• MICROFILM OVERLAP AREA •

40

39

38

37

SYM

2

REF PLANE II

REF PLANE I

3.81

PLY P28A
PLY P6A
PLY P5A
PLY P4A

PLY P1A

PLY P2A
PLY P3A

24

SYM

AA C

45°(TYP)

3.00(TYP)(REF)

PLY P28D

PLY P1D

CC

45°(TYP)

PLY P1

PLY P28

PLY P28

PLY P1

PLY P1B

PLY P28

PLY P1

AA D

PLY P28

PLY P28B

PLY P1

PLY P1B

SYM FOR PLYS P1A - P28A

3.81

AA C

45°(TYP)

PLY P28A

PLY P1A

3.00(TYP)(REF)

PLY P1

PLY P1B

45°(TYP)

PLY P1

PLY P28

PLY P28

PLY P1

PLY P1B

24

2

30.00

9.75(TYP)

4.88(TYP)

60.00

-03 LAMINATE DETAIL 2
VIEW LOOKING AT INNER SURFACE
SEE TABLE I

3

SR-013041 2

36

35

34

33

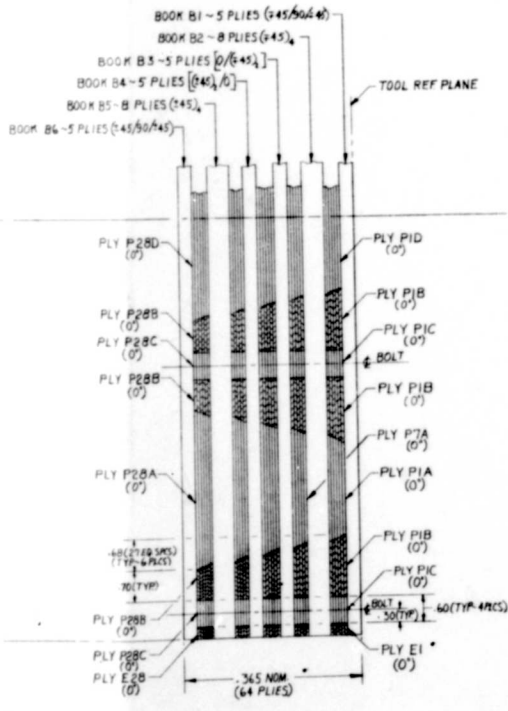
32

31

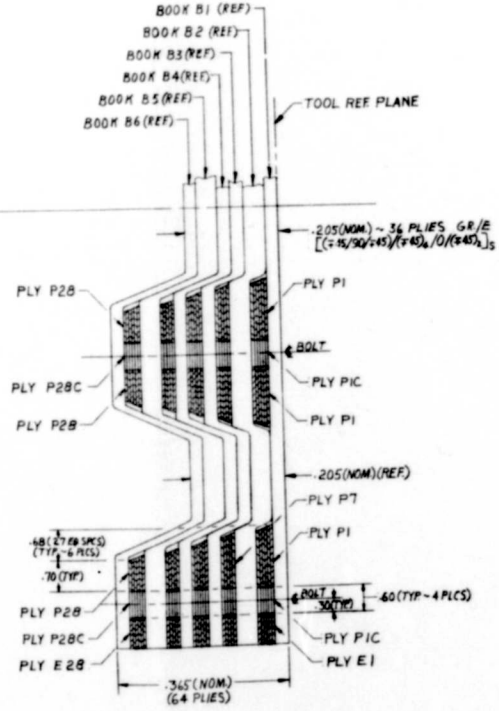
TEST PANEL REF PLANE

MULTI-DIRECTIONAL BOOKS			UNI-DIRECTIONAL PLYS		
BOOK No	PLY ORIENTATION	MTL	PLY Nos	PLY ORIENTATION	MTL
B1	(+45/0/+45)	GR/E A5/SSOI-5	E1-E2B	0°	GR/E SSOI-4B
B2	(+45)		P1-P2B		GR/E SSOI-4B
B3	[0/(+45)]		P7A-P2BA		GR/E A5/SSOI-5
B4	(+45/0)		P1B-P2BB		
B5	(+45)		P7C-P2BC		
B6	(+45/0/+45)	GR/E A5/SSOI-5	P1D-P2DB	0°	GR/E A5/SSOI-5

REVISION NOTES		REVISIONS		DATE	APPROVED
1	REFER TO SPECIFICATION DRAWING SET FOR PLY				



VIEW A
 SEE TABLE I
 SCALE: NONE



SECTION B
 SEE TABLE I
 SCALE: NONE

SR-013041

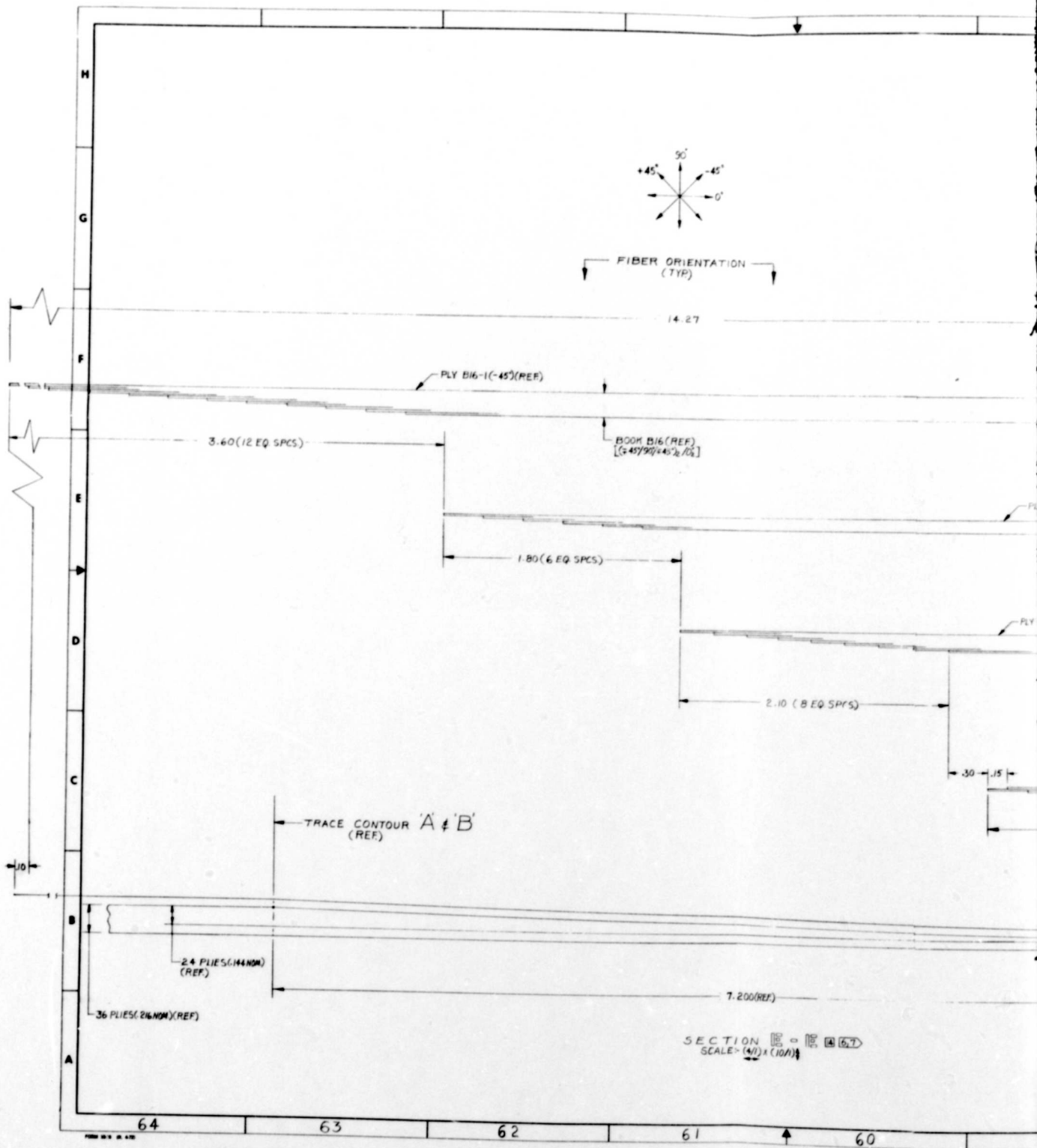
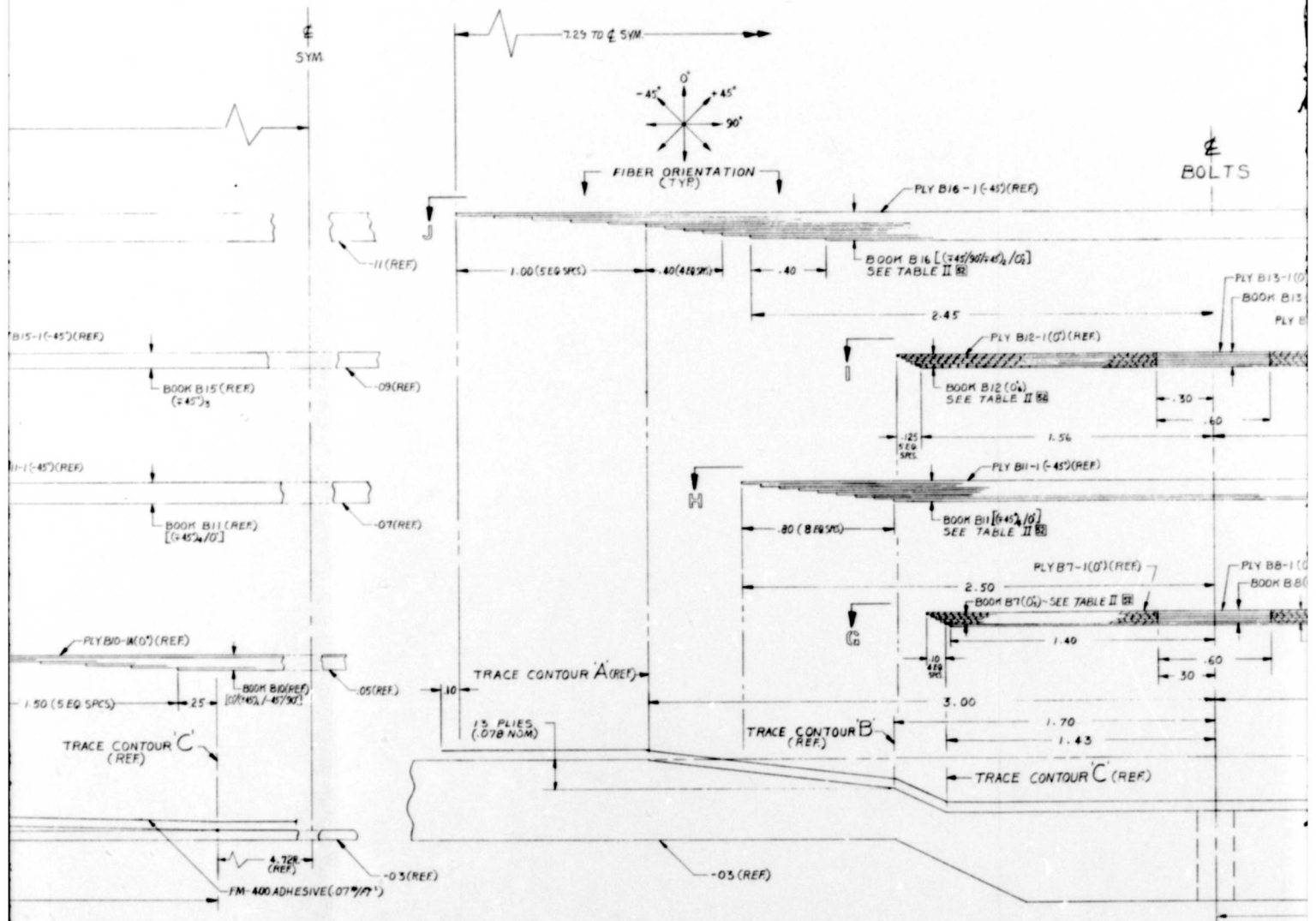


Figure 38. Panel Five Repair Drawing (Sheet 3 of 4)



SECTION [P] [P]
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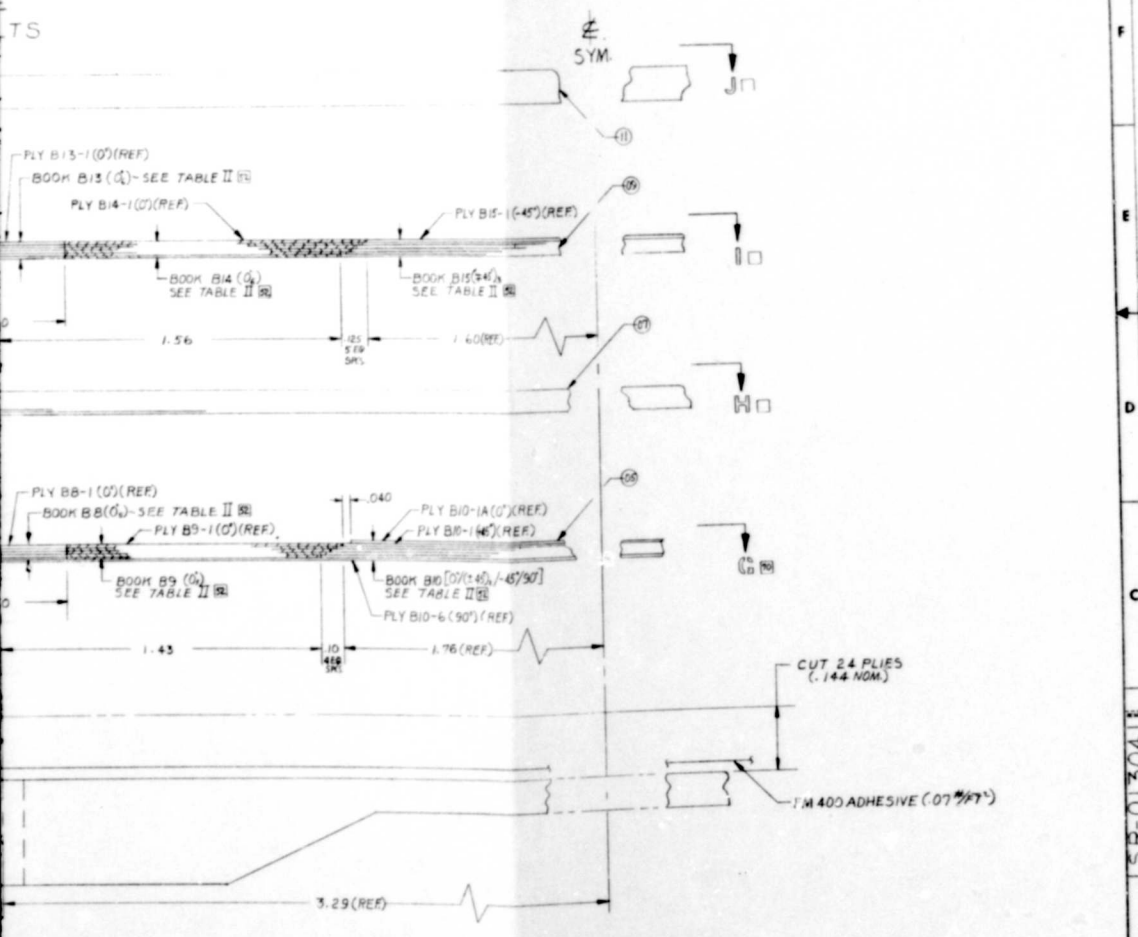
SR-013041 3

2

REVISION NOTES		REVISIONS			
NO.	DESCRIPTION	DATE	APPROVED	BY	DATE
1	REFER TO APPLICATION (SHEET) LAST FOR DET				

TABLE II

BOOK No	PLY ORIENTATION	MTL
B7	0°	80/W 5505-4B
B8	0°	84/W AS/5501-6
B9	0°	80/W 5505-4B
B10	0°/45°/45°	80/W AS/5501-6
B11	0°/45°/0°	84/W AS/5501-6
B12	0°	80/W 5505-4B
B13	0°	84/W AS/5501-6
B14	0°	80/W 5505-4B
B15	0°/45°	84/W AS/5501-6
B16	0°/45°/45°	80/W AS/5501-6



ON [R] [P] [M] [67]
 TATED 90° C.W.
 5/2" x (10x1) 1/2"

SR-013041	DATE ADDED	BY	DATE

53

52

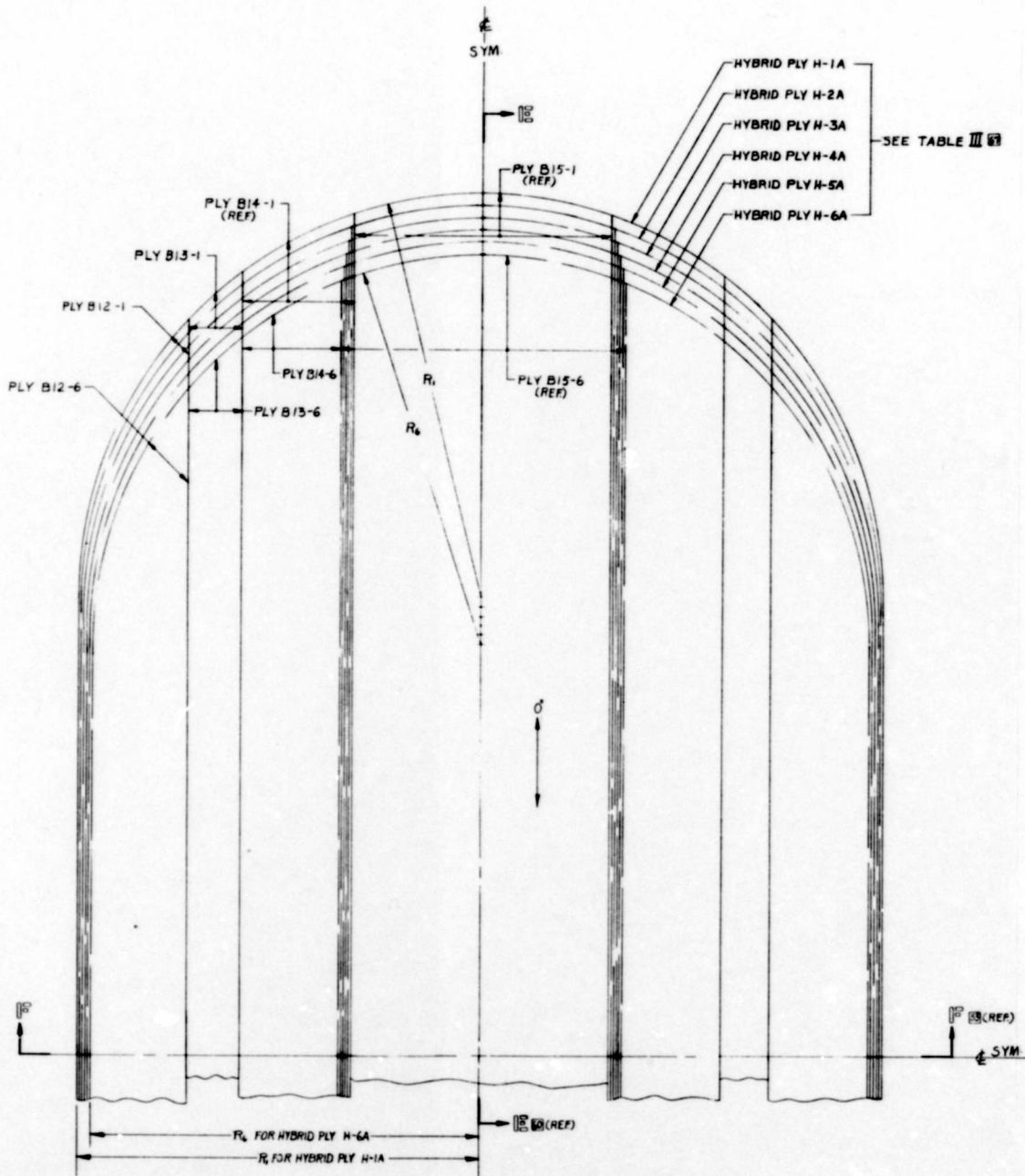
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49

3

SR-013041

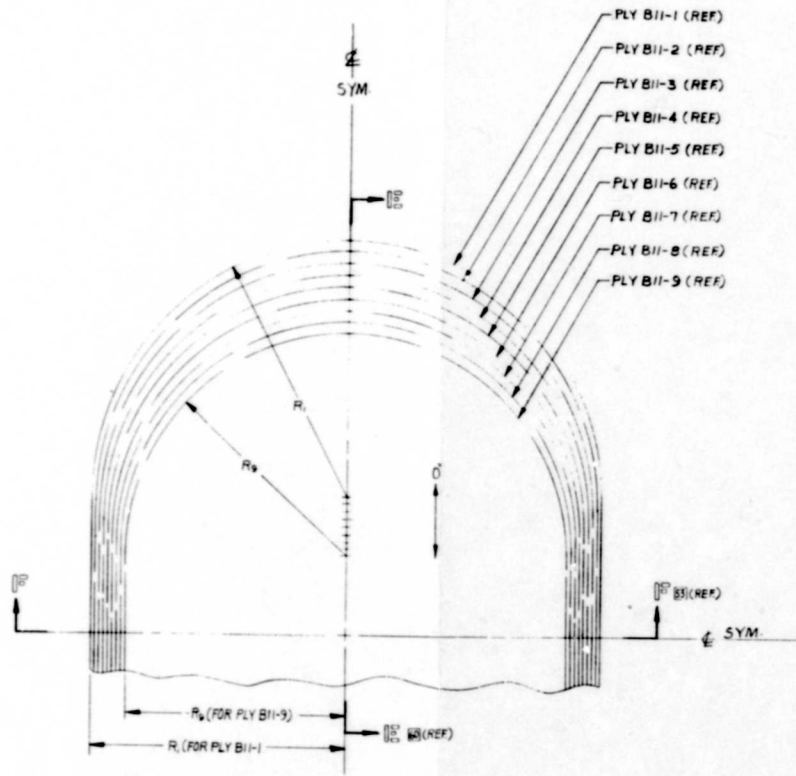


SEE TABLE III

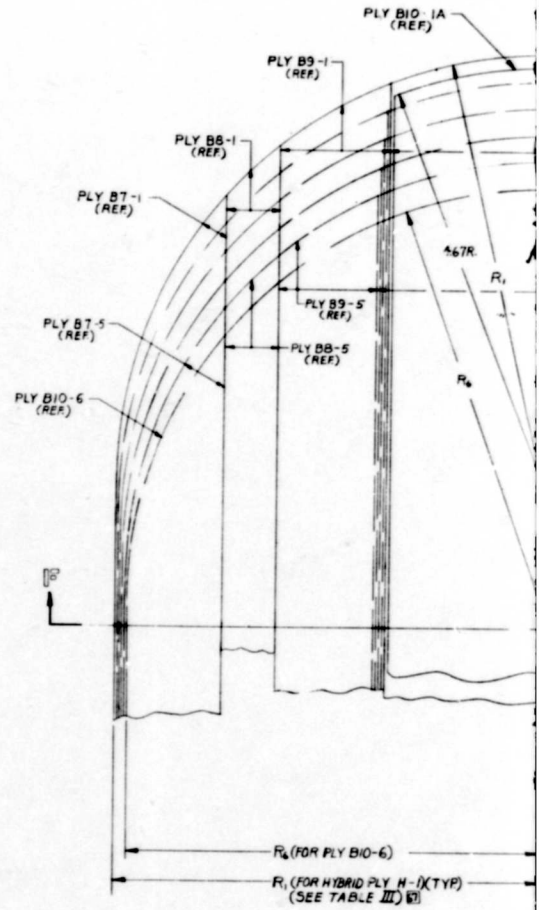
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SR-013041 4

2



VIEW H-H (FOR-D7 DETAIL)



VIEW (FOR S)

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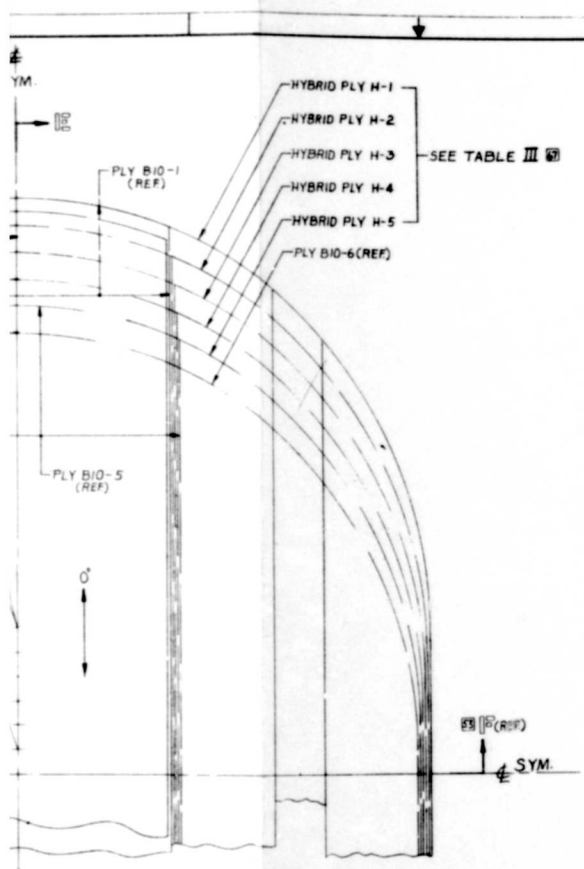
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3



REVISION NOTES		REVISIONS		
NO.	DESCRIPTION	DATE	APPROVED	
1	ADD TO SPECIFICATION (SEE SEE FOR DET)			

TABLE III (REF)

HYBRID PLY No	MAKE UP OF PLYS
H-1	B7-1 (2 REPS) B8-1 (2 REPS) B9-1 (2 REPS) B10-1 (2 REPS)
H-2	B7-2 (2 REPS) B8-2 (2 REPS) B9-2 (2 REPS) B10-2 (2 REPS)
H-3	B7-3 (2 REPS) B8-3 (2 REPS) B9-3 (2 REPS) B10-3 (2 REPS)
H-4	B7-4 (2 REPS) B8-4 (2 REPS) B9-4 (2 REPS) B10-4 (2 REPS)
H-5	B7-5 (2 REPS) B8-5 (2 REPS) B9-5 (2 REPS) B10-5 (2 REPS)
H-1A	B12-1 (2 REPS) B13-1 (2 REPS) B14-1 (2 REPS) B15-1 (2 REPS)
H-2A	B12-2 (2 REPS) B13-2 (2 REPS) B14-2 (2 REPS) B15-2 (2 REPS)
H-3A	B12-3 (2 REPS) B13-3 (2 REPS) B14-3 (2 REPS) B15-3 (2 REPS)
H-4A	B12-4 (2 REPS) B13-4 (2 REPS) B14-4 (2 REPS) B15-4 (2 REPS)
H-5A	B12-5 (2 REPS) B13-5 (2 REPS) B14-5 (2 REPS) B15-5 (2 REPS)
H-6A	B12-6 (2 REPS) B13-6 (2 REPS) B14-6 (2 REPS) B15-6 (2 REPS)

W (G) = (G) (REF)
 1-05 DETAIL
 SCALE = 2, 1

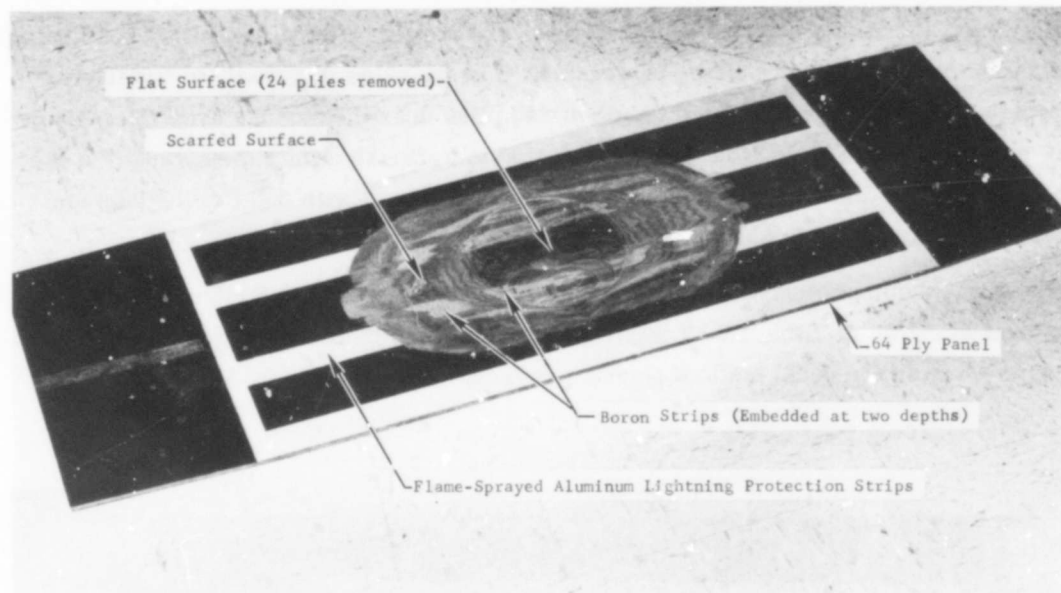
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70 69 68 67 66 65

4

M
G
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C
A

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78-04301-3

Figure 39. Panel Five After Material Removal

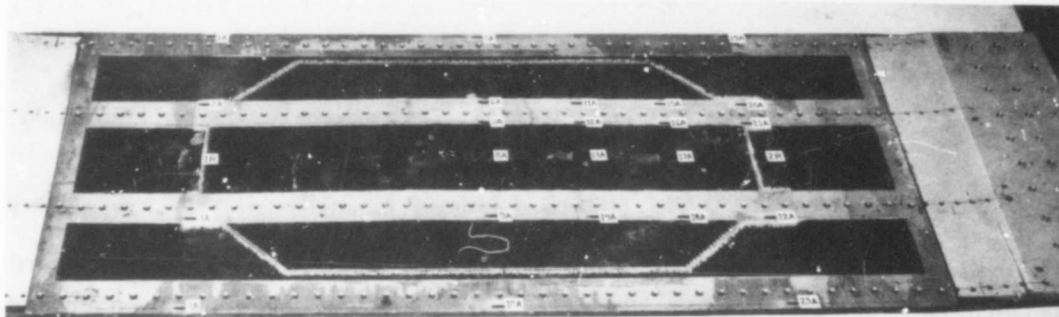
and cutting along the lines, starting with the largest piece first and working in toward the smallest, each ply was cut to the desired size. The plies were laid up on a paper print made from the mylar, starting with the smallest and working out to the largest, so that the pattern of the print located 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 inch thick to replace the lightning protection system in the repair area. After all of 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° orientation, the distortion was not expected to seriously affect the strength in the 0° primary load direction. The completely repaired panel with end loading tabs and strain gages installed is shown in Figure 40.

The Grumman Aerospace Corporation supplied information for the four limit load cases shown in Table 15. Longitudinal loading was defined by strain while shear flow is given in the usual units of pounds per inch.



78-05888

Figure 40. Strain Gage Installation of Test Panel Number Five

TABLE 15. LIMIT LOADS FOR B-1 HORIZONTAL STABILIZER COVERS⁽¹⁴⁾

Limit Load Condition	ϵ_x $\mu\text{In.}/\text{In.}$	N_{xy} $\text{Lb.}/\text{In.}$
(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

14. Herman Erbacher, Letter AD C PM 78 195, Grumman Aerospace Corp., Advanced Composites Group, Sept. 1978.

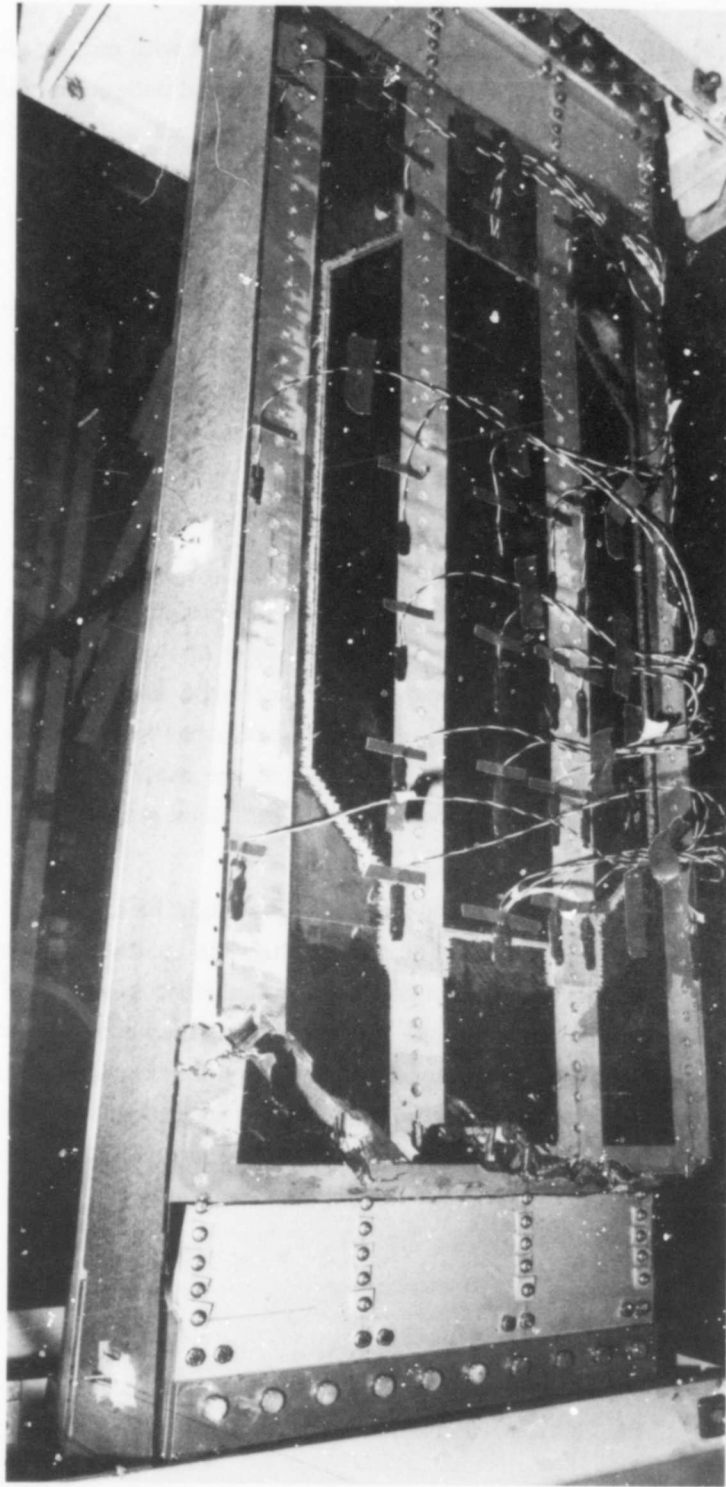
The four limit load conditions of Table 15 were applied to the panel with correction for dead weight bending and ignoring the sign on N_{xy} . Load, strain and deflection measurements were taken at 0, 30, 50, 70, 90 and 100 percent loads. No indication of failure was observed. Four ultimate load conditions were then applied to panel five using 150 percent of the values of Table 15 to define the loads. No indication of failure was observed using visual, auditory and coin tap inspection.

The failure load condition imposed uniform bending on the test section of the box resulting in tension in the hybrid cover. Failure occurred at a total moment of 2,490,000 pound inches (including dead weight moment of 46,500 lb. in.) for an average cover load intensity of 19,300 lb./in. Failure occurred remote from the repair as shown in Figure 41. The ultimate allowable loading was defined by Grumman Aerospace Corporation,⁽¹⁵⁾ as that load resulting in a strain ϵ_x allow. = 4171 μ in./in., which is 80 percent of the failing strain prediction value.

The strain gage data was found to be linear with load, essentially to failure. Strains at failure (scaled from the last value read) are shown in Figure 42. On the left end, the distribution is linear from edge to center. Therefore, the mean strain between the edge and the center is taken as the value to be compared to the strain allowable. This mean failure strain is 6300 μ in./in. Hence, the panel failure occurred at 151 percent of the design ultimate allowable. Maximum strain in the parent laminate was 7912 μ in./in.

The strains in Figure 42 are not doubly symmetrical as might be expected. Since the strain distribution remains the same during loading, the spatial strain variations are due to stiffness variations, not local failures. Local fiber waviness is suggested as the cause of the stiffness variation. In determining the maximum strain at a fastener hole, only the gages outside the patch were used since the high strains on gages 6A and 7A were considered to be unusually high, but did not influence the failure. Hence, the maximum strain level at a hole was 6900 μ in./in. at failure interpolated between gages 1R2 and 3A.

¹⁵ Les Murrin, Letter AD C PM 79 16, Grumman Aerospace Corp., Advanced Composites Group, January 1979.



78-06136

Figure 41. Failure of Test Panel Number Five at 151% of Design Ultimate Allowable

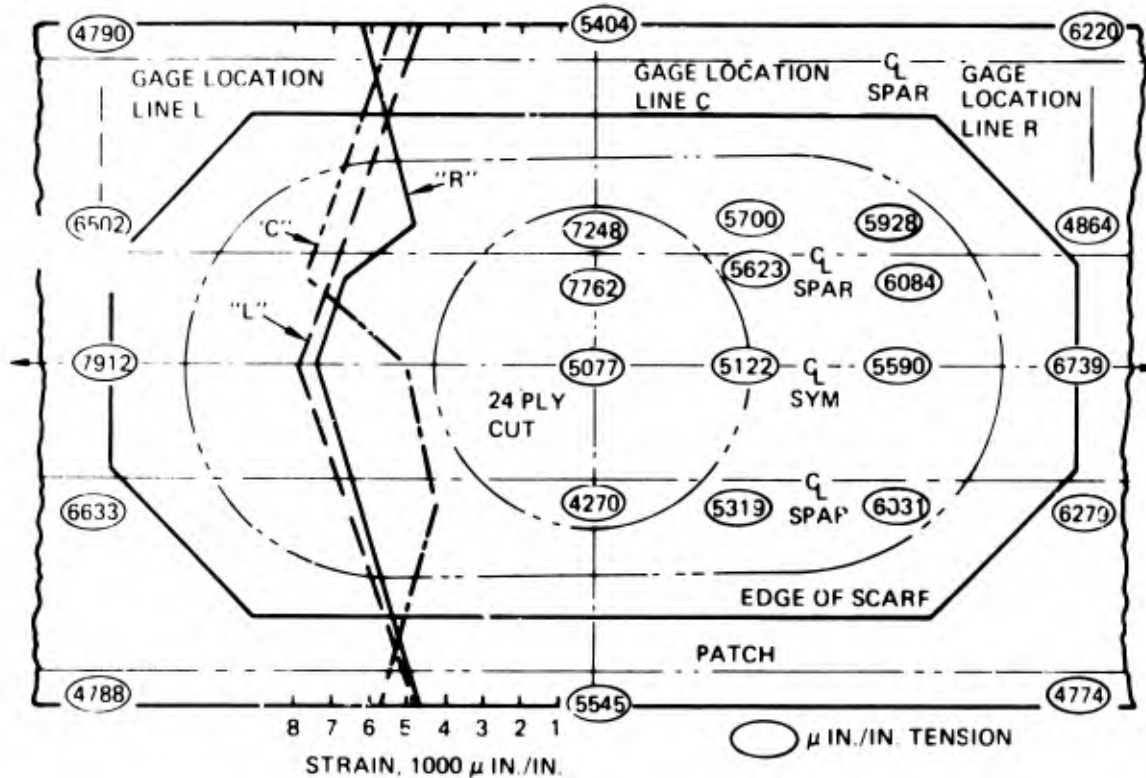


Figure 42. Failure Strains on Panel No. 5

4.6 SUMMARY OF TEST PANEL RESULTS

A summary of the test results of Panels One through Five is shown in Figure 43, 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 graphite structure to its original strength which is well above current design ultimate allowables.

4.7 F-15 SPEEDBRAKE REPAIR

Several speedbrakes for F-15 aircraft were fabricated of T300/5208 graphite epoxy with an aluminum honeycomb core.⁽¹⁶⁾ One of these was damaged, reportedly

¹⁶M. N. Botkin, B. J. Colvin, W. B. Finuf, and R. B. Kollmansberger, F-15 Composite Speedbrake, Air Force Flight Dynamics Laboratory, Wright Patterson AFB, Ohio, 45433, AFFDL-TR-75-133, Nov. 1975 (McDonnell Aircraft Co.).

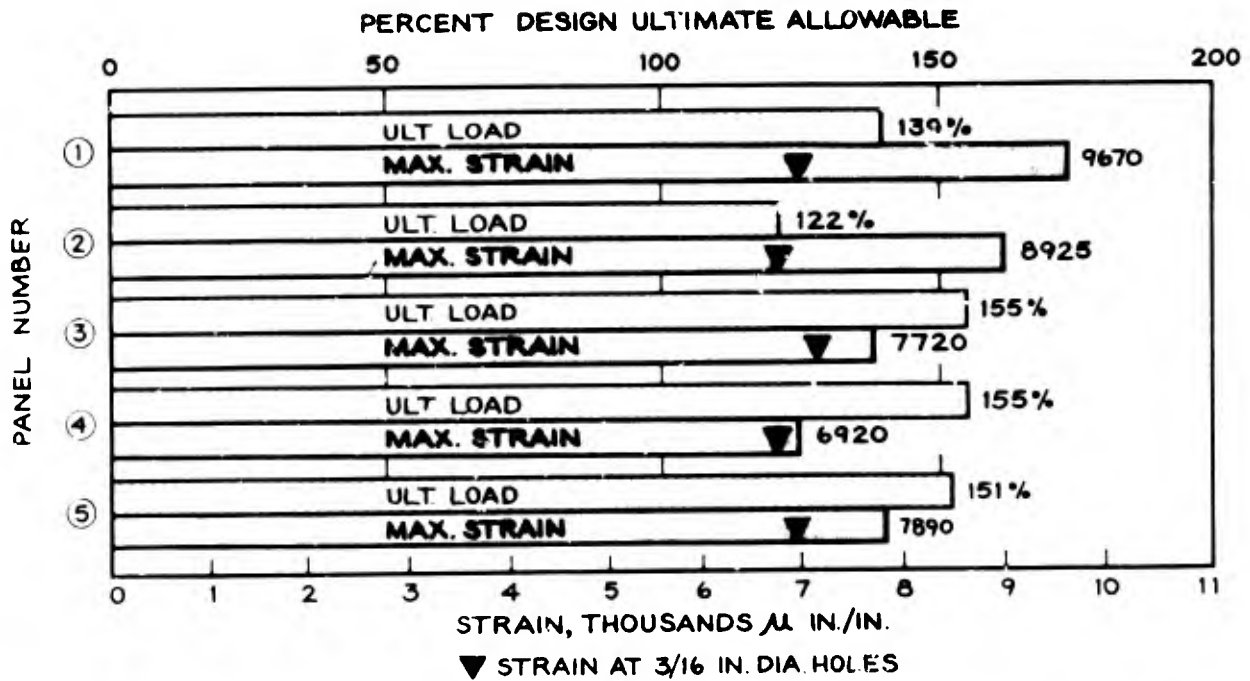
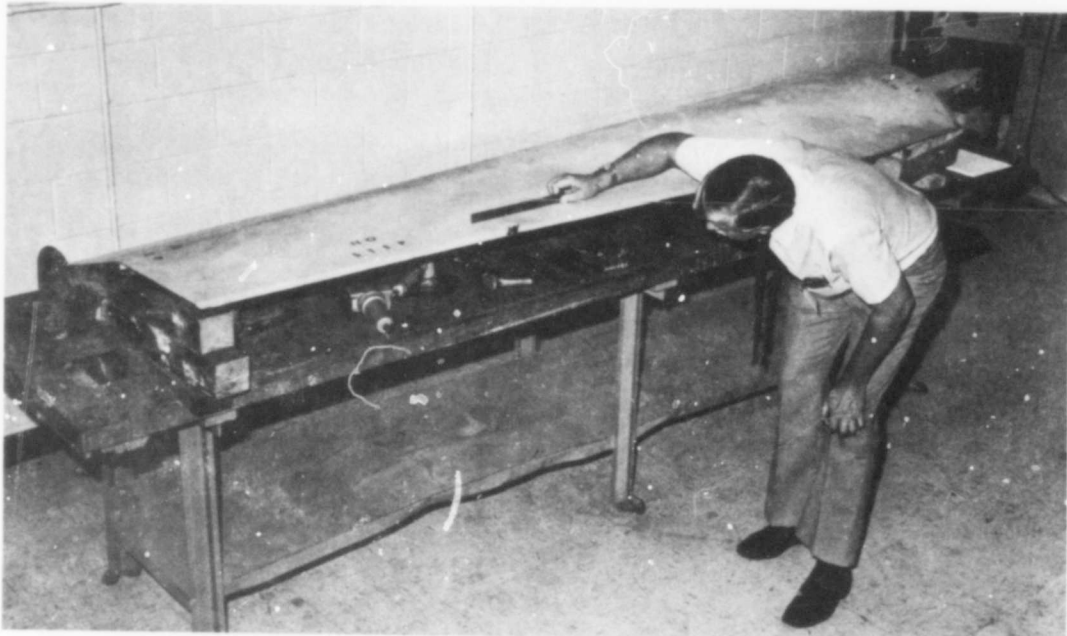


Figure 43. Summary of Demonstration Panel Results

when a canopy came off in flight and impacted the speed brake. The damaged speed-brake was made available to this program for use in demonstrating the application of the repair procedures on a real aircraft part. A flush repair was made and the speedbrake was proof tested to verify the quality of the repair.

The speedbrake is approximately 10 feet long and 3 1/2 feet wide and contains machined titanium actuator fittings and hinge arms which are integrally bonded with the graphite/epoxy skins and aluminum honeycomb core. The damage consisted of an indentation on the outer face approximately 6 inches by 7 inches in area and 1/4 inch deep, located near the edge of the part about 3 feet forward of the aft end as shown in Figure 44. 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 which could be used for on-aircraft repairs. A hand held router was used to cut the 5 to 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. The router was used to remove strips of core so that the remaining pieces were easily removed with a knife.

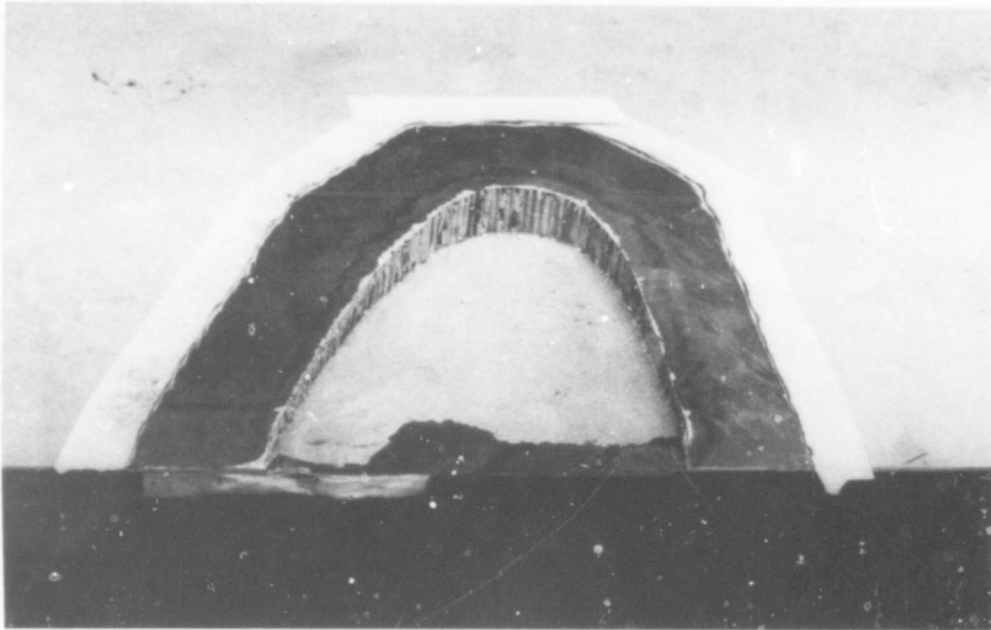


78-05561-7

Figure 44. Damaged F-15 Speed Brake

At the edge of the opening which had been cut in the outer skin, the skin was scarfed over a width of less than an inch using a hand held air-driven motor with a 30-inch sanding disk, and a small belt sander. The scarfed surface was shaped to end at a scribed line on the surface. The hand held sander was controlled by the operator without using any mechanical guides. The repair area with the core removed and the scarf surface prepared is shown in Figure 45.

A heat blanket was used for curing the repair on the F-15 speedbrake in order to demonstrate a technique which might be used for on-aircraft repairs. A preliminary verification of the technique was made by curing an 8-inch by 10-inch, 7 ply panel using vacuum and a heating blanket. The heat was controlled by monitoring a thermocouple on the laminate and manually adjusting the current flow to the blanket. An 0.020 inch 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 were placed over the blanket. Four thermocouples were used to verify the temperature at four locations on the part. At three points, the temperatures were satisfactory and uniform within 10°F. At one edge, having only 3/4 inch of blanket extending beyond the edge of the part, the temperature



76-06465

Figure 45. Repair Area on F-15 Speed Brake

was consistently too low. At the opposite edge, the blanket extended 2-inches beyond the part and temperatures were essentially the same as the control point which had blanket extending 13-inches beyond the part. To simulate the heat sink characteristics of the actual speedbrake, the 8 x 10 inch test panel was cured on a portion of one of the large scale box beam covers having a honeycomb core. The panel was cured for 2 hours at 350°F under vacuum bag pressure.

A similar panel cured in an oven with vacuum pressure was used as a basis of comparison for strength coupons, physical property measurements and photomicrographs. In each comparison the panel cured with the heat blanket was of equal or better quality than the oven cured panel. Based on these results, the blanket cure procedure was used for subsequent curing of the repair on the 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.

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 (at a slope $L/t = 36/1$ for this repair), a layer of FM 400 film adhesive, 5 plies of AS/3501-6 undirectional tape prepreg replacing parent

laminated plies (plus a zero degree 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 end of outer plies. Paint was removed in the repair area by light hand sanding with fine sandpaper.

The size and shape of the replacement plies and the added surface plies were defined by drawing a pattern on a sheet of semi-transparent nylon film by tracing the location of the edges of each ply on the scarf surface and allowing for 0.10 inch overlaps. Two copies of the pattern were made, one to use to cut the plies and the second for locating them as they were stacked prior to their location on the part.

The template used to define and cut the upper surface patch plies is shown (reduced from full size) in Figure 46.

A small area of the inner skin approximately 3.0 x 1.5 inches 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 tap examination and was injected with Epon 815 epoxy, using the procedure previously shown to be satisfactory for Type (3) specimens for which blisters also occurred, as described in Paragraph 3.1. The template for the inner (lower) surface patch plies is shown (reduced from full size) in Figure 46. 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 Figure 47. In order to obtain a more severe test of the repaired area, unsymmetrical loading condition No. 1 shown in Figure 48 (from reference 16) was applied, reversed to place the larger loading pressure on the same side as the repair. Loading pads were located aft of the repair, but immediately adjacent to the repair to ensure that bending and torsion loads would be carried through the repaired area.

Fifteen strain gages were used to monitor the test loading. Gage locations and strain readings at 120 percent of the limit loading shown in Figure 48 are shown in Figure 49. The peak strain shown is 5583 $\mu\text{in./in.}$ adjacent to the patch. No damage to the repaired area occurred and proof loading was terminated at 120 percent of the limit loading condition, after demonstrating that the repair was adequate.

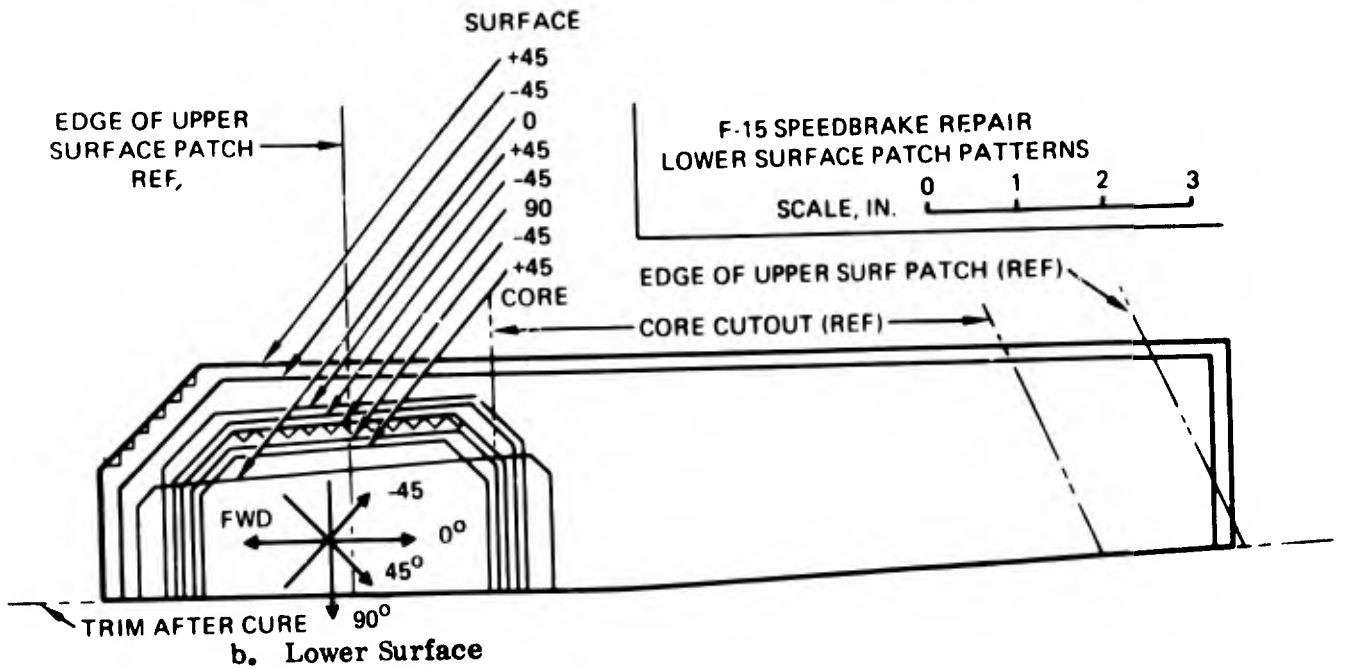
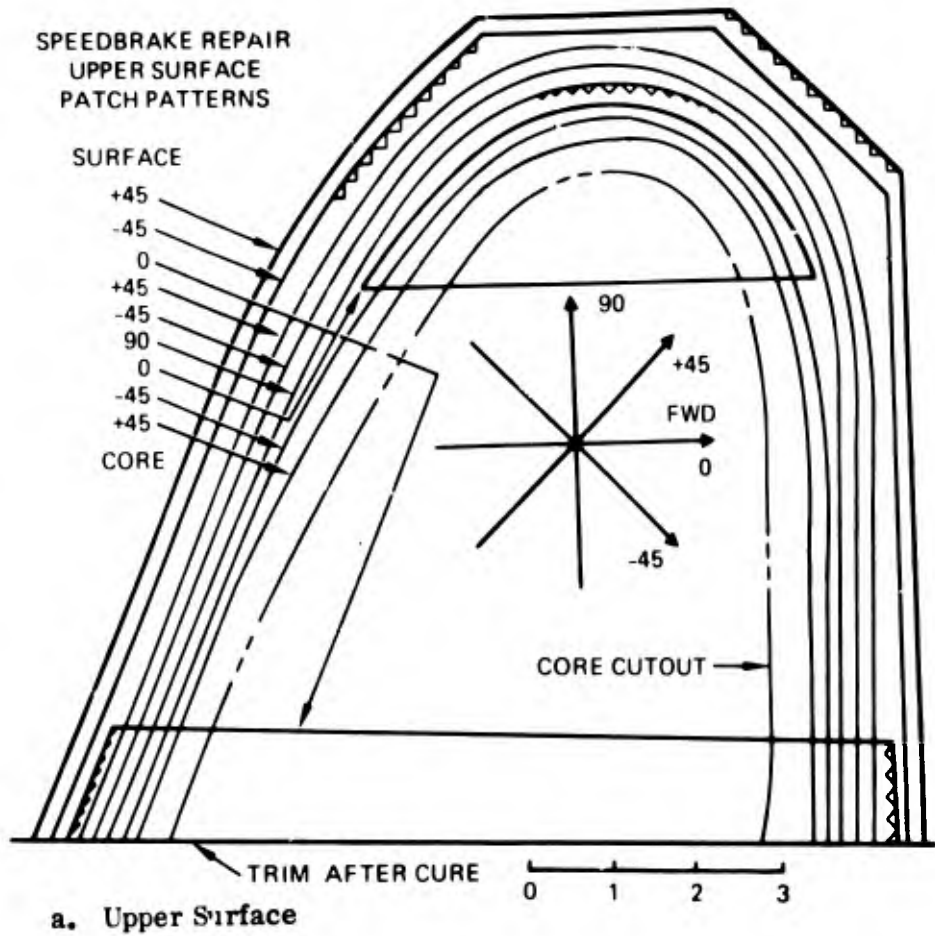
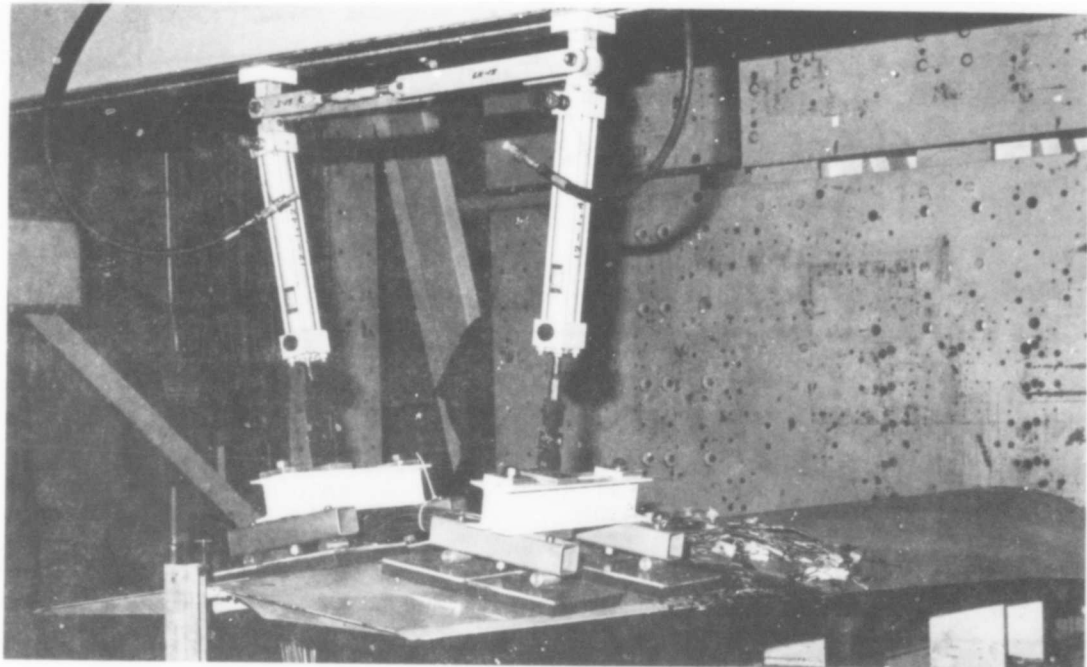


Figure 46. Repair Ply Patterns for F-15 Speedbrake



79-00233-3

Figure 47. F-15 Speed Brake Test Arrangement

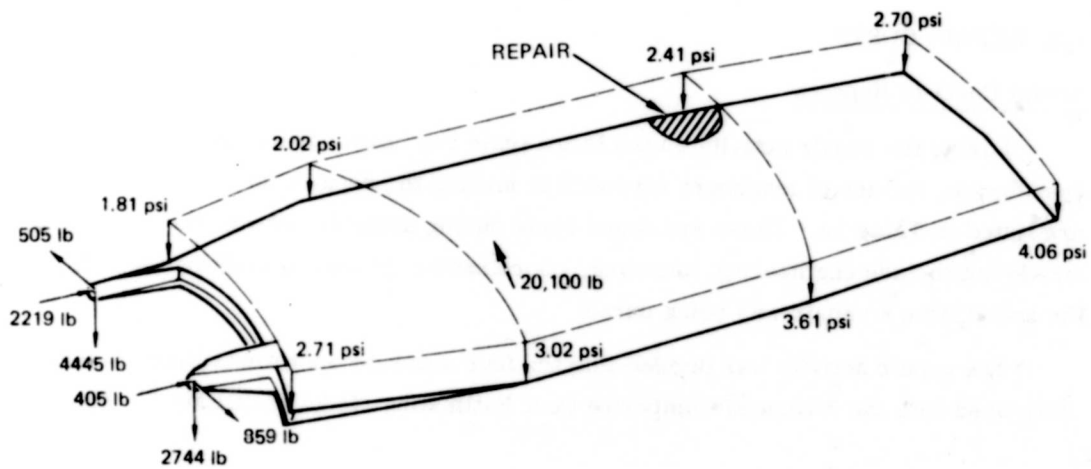


Figure 48. Proof Test Limit Load Case

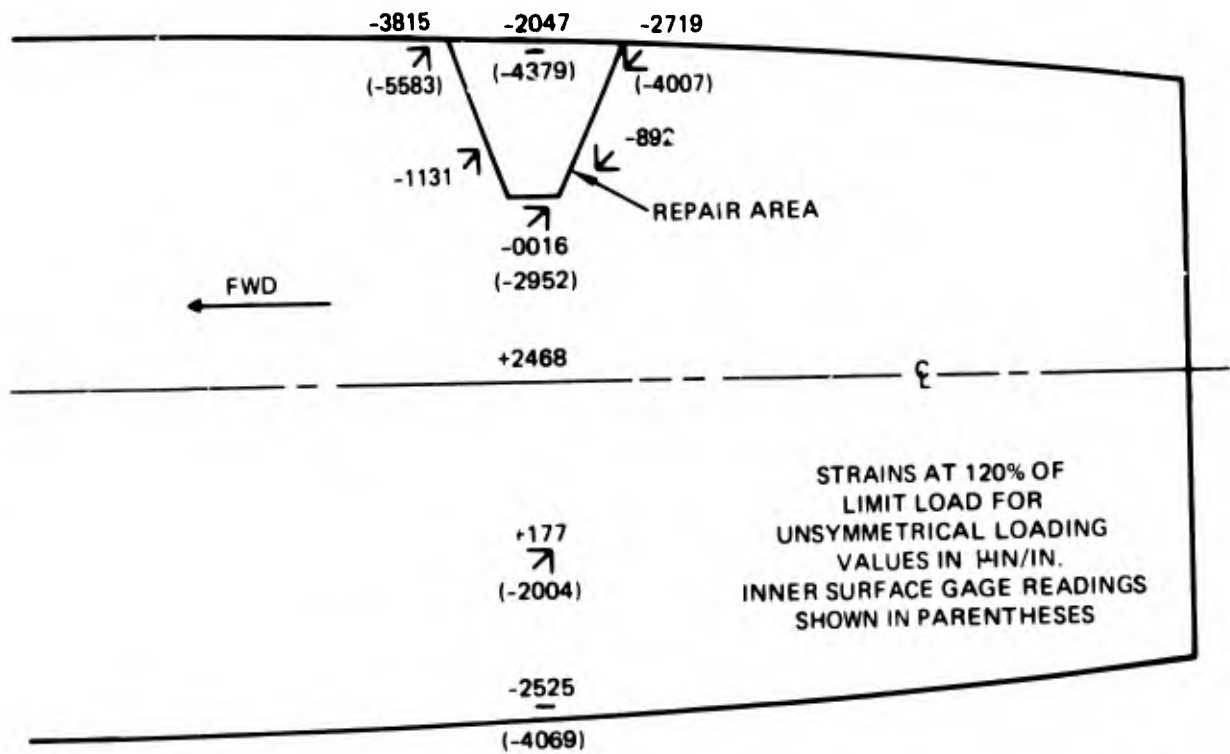


Figure 49. Strains at 120% of Limit Load During F-15 Speed Brake Proof Loading

Motion picture film was made of each operation during the repair and proof test of the speedbrake. The film demonstrates the techniques as used for a realistic application of the repair concepts developed in this program.

4.8 REPAIR COSTS

Actual Costs of Repairs

During the repair activity on the large scale box beam covers and on the F-15 speedbrake, the actual manhours expended in making the repairs were recorded, and are listed in Table 16. These are times spent during actual hands-on shop work only, and do not include engineering, planning, coordination, preparation of tooling aids, or the actual time a repair was being cured.

The repair activity was divided into the four separate operations listed to match those used with the Advanced Composite Cost Estimating Model (ACCEM).¹⁷ The

¹⁷Advanced Composites Cost Estimating Model, Air Force Flight Dynamics Laboratory, Wright Patterson AFB, Ohio, 45433, AFFDL-TR-75-3103, Aug. 1976, (Northrop Corp.)

TABLE 16. ACTUAL REPAIR HOURS EXPENDED

OPERATION	HONEYCOMB SANDWICH PANEL		SOLID LAMINATE PANEL		HYBRID REPAIR ON ONE SIDE	F-15 SPEED-BRAKE
	TWO SIDES ACCESSIBLE	ONE SIDE ACCESSIBLE	REPAIR ON ONE SIDE	REPAIR ON TWO SIDES		
LAYUP	6.28	7.90	4.76	14.65	15.86	9.27
CORE PREPARATION	0.46	1.76	0	0	0	0.89
PART CONSOLIDATION	1.00	8.50	0.53	0.95	0.75	1.75
FINISHING/MACHINING	9.80	9.27	11.55	5.60	8.60	3.20
TOTAL HOURS	17.54	27.43	16.84	21.20	25.21	15.20

model uses factory standard labor hours for each of these operations during the fabrication of a part to estimate fabrication costs. No data is available for standard hours for repair activity, so factory standards were used to estimate repair costs in this program. Detailed examples were given in the third interim progress report, and results are summarized here.

The standard hours for each operation during the fabrication of a part are grouped into four categories: layup, honeycomb core operations, part consolidation and finishing. The layup includes depositing the composite material, ply handling, cutting, debulking, and draping over contoured tools. Aluminum honeycomb core operations include raw stock sawing, machining, core forming and stabilization. Part consolidation includes vacuum bagging; activity associated with autoclave, oven, and elastomeric curing (but not time during the cure); bonding of detail parts; and splicing of core elements. Finishing encompasses all trimming and drilling operations performed on a cured part.

The standard hours are the idealized, minimum time required to perform the operation. To account for realistic delays, variances are applied to the standard hours based on previous experience, to obtain direct labor hours. Allowances cover waiting time for tools or equipment, attention to personal needs, time lost to fatigue, coffee breaks, work planning, clean up, etc. Support functions are also estimated covering labor support such as engineering, tooling, manufacturing engineering, quality control and graphics services. Production material costs are also computed with allowance made for scrap. Other material, such as curing bags, are separately included. The weight of the part is also computed.

Predictions using this model account for the previous experience or the "learning curve." The times predicted for the first repair, the tenth, twentieth, etc., shows that with increasing experience, the time required reduces rapidly. For example, predicted repair costs for the first sandwich panel box beam cover with both sides accessible are approximately 81 hours for the first repair, 37 hours for the tenth, and 20 hours for the 50th, based on the factory standards. The actual time was 17.54 hours, corresponding to an experience level of over 50 repairs. The personnel making the repairs had been associated with the program between one and two years and had considerable previous related experience. Therefore, the data tend to indicate that the predictions are reasonable for repair activity by experienced personnel, but that inexperienced personnel making their first repairs may require many times the hours listed in Table 16.

Repair vs. Replace Considerations

Cost is one factor which can influence the decision of whether to repair or to replace a component. This decision obviously will also be influenced by spares availability, aircraft down time, personnel skills, shop work load, and availability of necessary repair materials and equipment. When these other factors do not govern the decision, however, the relative costs of making a repair vs. scrapping the part and replacing it become of interest.

During the fabrication of the five large scale box beam covers, the ACCEM was used to estimate fabrication costs, using factory labor standard hours, variance factors to account for normal delays, and improvement curves for the influence of experience. A comparison can therefore be made between repair time and fabrication time. In making such a comparison, the relative simplicity of the box beam covers must be considered as compared with an actual aircraft component such as a flap, a rudder, etc. These panels did not contain embedded attachment fittings, access cutouts, hinges, etc., and therefore their fabrication costs are low compared to most typical components. The relative costs of fabrication of a replacement panel are compared with the repair of the damaged panel in Table 17. Since replacement parts are assumed to be fabricated during factory production, costs are based on experience factors for unit 300, while repair costs assume the 50th repair, i.e., experienced repair personnel. Non-direct costs of production, spares provisioning, etc., are not included, but would probably add substantially to the cost of a replacement part and make the repair more cost effective.

TABLE 17. REPAIR VERSUS REPLACE COST COMPARISONS

LARGE SCALE BOX BEAM COVER	FABRICATION ¹ COST OF REPLACEMENT	REPAIR COST ²
1. Sandwich-Accessible 2 Sides		
Factory (or Repair) Labor	\$ 208	\$ 197
Production (or Repair) Material	\$ <u>171</u>	\$ <u>33</u>
Total	\$ 379	\$ 230
2. Sandwich-Accessible 1 Side		
Factory (or Repair) Labor	\$ 208	\$ 324
Production (or Repair) Material	\$ <u>171</u>	\$ <u>23</u>
Total	\$ 379	\$ 347
3. Solid Panel-Repair One Side		
Factory (or Repair) Labor	\$ 262	\$ 188
Production (or Repair) Material	\$ <u>480</u>	\$ <u>42</u>
Total	\$ 742	\$ 230
4. Solid Panel-Repair Two Sides		
Factory (or Repair) Labor	\$ 262	\$ 211
Production (or Repair) Material	\$ <u>480</u>	\$ <u>52</u>
Total	\$ 742	\$ 263
5. Hybrid Panel-Repair One Side		
Factory (or Repair) Labor	\$ 788	\$ 240
Production (or Repair)	\$ <u>1,099</u>	\$ <u>57</u>
Total	\$ 1,887	\$ 297
1 Based on production of unit 300		
2 Based on 50th repair of similar panels		

The result of the cost comparison shows that with the possible exception of very simple panels which are readily available as spares, it will normally be more economical to repair damage than to replace the component, especially after some experience has been developed with repair procedures.

SECTION V

CONCLUSIONS

1. Graphite/epoxy structure can be repaired to restore between 80 and 100 percent of the parent laminate allowable strength.

Extensive testing has shown this strength restoration to be possible for temperatures from -65° to 265°F , for parent laminates containing 1 percent moisture, for repairs reexposed to moisture, and for specimens exposed to fatigue loading. Repairs made on 5 realistic, large scale panels consistently restored their original strength by developing strains from 6900 to nearly 10,000 $\mu\text{in}/\text{in}$.

2. Repairs have been shown to be permanent and durable for the life of the aircraft.

Exposure of repaired panels to 2 lifetimes of random spectrum fatigue loading with a peak load causing 4570 $\mu\text{in}/\text{in}$ strain has shown no significant degradation of strength. Static tests have shown that exposure to hydraulic fluid, fuel and paint stripper has less effect than exposure to moisture, and repairs were repeatedly shown to retain over 80 percent of the parent laminate allowable strength in the presence of moisture.

3. Repairs can be made satisfactorily in areas where severe criteria apply for aerodynamic smoothness or thermal loading.

A variety of repair configurations were developed for which aerodynamic smoothness criteria were met by limiting the out-of-moldline buildup to less than 0.020 inch total, with tapered edges. Adhesives were used with a nominal 350°F service temperature, and extensive testing of repairs was successfully conducted at 265°F .

4. Repairs to damaged areas with no limit on size can be made at the Air Force depot repair facilities, i. e., the Air Logistics Centers (AIC's).

The techniques developed do not require specialized tools or equipment, although most personnel will require some training and experience. Techniques are available for either on- or off-aircraft repairs, including situations in which access is available only to one side of the panel. Techniques have been demonstrated for which vacuum bag curing is acceptable without autoclave pressure. Portable heat blankets have been successfully used. Hand tools have been shown to be acceptable for damage removal and surface preparation. Although developed specifically for depot level activity, many techniques may also be applicable at the organization and intermediate (O&I) levels.

5. Neither the design nor the implementation of repair techniques are difficult, although several details of the techniques need to be understood by the personnel involved.

Stacking sequence of patch plies can affect the strength of the repair, especially for plies near the outer surface. The tendency for surface plies to peel can be reduced by serrating their ends on cuts normal to the fiber direction. Minimum overlap lengths are required to develop the strength of each patch ply. A potential lower strength for patch material, which can result from curing parameters, can easily be compensated for by the selective addition of extra load carrying plies. An external patch repair configuration was found to restore less total strength than the modified scarf joint, to be more difficult to design, and to offer little, if any, saving in effort required. Techniques have been developed for making repairs with access to only one side of the damaged panel. If not considered during the processing, a high moisture content can cause blistering of the parent material during high temperature curing of the repair. Use of a low resin content prepreg for patch plies can eliminate the need for resin bleed during cure.

SECTION VI
RECOMMENDATIONS

1. Techniques for repair of advanced composite substructure should be developed.

The cocuring approach using bonded joints which was applied to skin panels in this program offers possible advantages for substructure repair. These include accessibility requirements for fit-up of prefabricated parts, and increased strength by eliminating mechanical fasteners from the repair.

2. Maintenance personnel should receive training applicable to advanced composite structural repair.

Coordination meetings with ALC personnel during this program have indicated a need for specific instruction for both engineering and shop personnel. Although not difficult, repair techniques for advanced composites differ from either standard metal or fiberglass methods. Ideally, personnel should develop experience through experimentation prior to making actual repairs on real aircraft structure where safety of flight is involved.

3. A method of measuring the moisture content of thick laminates is urgently needed.

Because of the potential for blistering of the parent laminate during high temperature repair cure operations when moisture has been absorbed, it is necessary to estimate the moisture content and its distribution in order to dry the material, to reduce the cure temperature, or both, to prevent further damage. Techniques are currently being investigated which use both dielectric measurements and the acoustic emission resulting from a limited temperature rise.

4. Additional data are needed to prevent blistering of wet parent material during cure of repair plies.

A parametric investigation is recommended to isolate the effect on blistering of repair ply cure temperature, parent laminate thickness, moisture content,

moisture distribution through the thickness, glass transition temperature and the source of the tendency to blister, i. e. , contamination, etc. Such an investigation should be intended to provide simplified direction to repair personnel to prevent blistering by varying repair ply curing procedures, and should therefore be related to "typical" and "worst case" environmental exposures to be expected in service.

5. Better definition is needed for the extent and type of repair which is required.

The effects of minor damage on strength need to be established to assist maintenance engineering personnel in the selection of the repair procedures to be used. In many cases a quick, cosmetic repair may be acceptable, while at other times damage which appears negligible may actually cause unacceptable strength loss so that structural repair is required. This requires application of NDI techniques to define the nature and extent of damage, and development of data which will specify the effect of the damage on strength. Work is currently in progress in both of these fields.

6. Procedures for application of nondestructive inspection methods to repairs are needed.

NDI methods can be applied to the identification of damage to define the extent of the repair needed, and can also be used after the repair is completed to verify its quality.

7. Simple repairs which can restore high strength for limited damage areas are needed for Organizational and Intermediate (O&I) levels of maintenance.

Techniques which show promise include room temperature curing materials, ultraviolet light activated materials, two part resins, precured materials and configurations which minimize damage removal.

8. Repair procedures should be consolidated into a handbook for easy reference.

The procedures for repair of composite materials which have been developed for a variety of applications in this and other research programs should be reviewed and combined with recommendations for their use into a single document for reference use by service maintenance personnel.

REFERENCES

1. G. Lubin, et al., Repair Technology for Boron/Epoxy Composites, Air Force Materials Laboratory, Wright-Patterson Air Force Base, Ohio 45433, AFML-TR-71-270, February 1972. (Grumman Aerospace Corp.)
2. R. M. LaSalle, Repair Procedures for Advanced Composite Structures, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-76-57, December, 1976. (General Dynamics Corp.)
3. James C. Watson, Field Repair of Composite Structures, Naval Air Development Center, Warminster, PA. (McDonnell Aircraft Co.)
4. A. L. Scow, et al., Large Area Composite Structure Repair - First Interim Progress Report, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-77-5, January 1977. (Northrop Corp.)
5. R. W. Kiger and S. H. Myhre, Large Area Composite Structure Repair - Second Interim Progress Report, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-77-121, November 1977. (Northrop Corp.)
6. R. W. Kiger and S. H. Myhre, Large Area Composite Structure Repair - Third Interim Progress Report, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-78-83, July 1978. (Northrop Corp.)
7. R. M. Verette and J. D. Labor, Structural Criteria for Advanced Composites, Final Report of Contract No. F33615-74-C-5782, Vol. 1, Air Force Flight Dynamics Laboratory, Wright-Patterson Air Force Base, Ohio, 45433, AFFDL-TR-76-142, March 1977. (Northrop Corp.)
8. Private Communication with R. L. Stoecklin, Boeing Commercial Airplane Co. regarding unpublished data on 2 year exposure data showing approximated 1% maximum moisture absorption for worldwide environments.
9. D. L. Reed, Point Stress Laminate Analysis, Report FZM-5494, General Dynamics/Fort Worth, April 1970.
10. C. H. Shen and G. S. Springer, Moisture Absorption and Desorption of Composite Materials, Journal of Composite Materials, Volume 10, 1976.
11. A. O. Kays, Determination of Moisture Content in Composites by Dielectric Measurements, Contract No. F33615-78-C-3216, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, 45433 (Lockheed-Georgia Co.)

12. D. E. Parker, Development of a Low Cost Composite Vertical Stabilizer, Contract No. F33615-74-C-5164, Second Quarterly Progress Report, Rockwell International, NA-75-99-2. July 1975.
13. Advanced Development of Conceptual Hardware, Air Force Materials Laboratory, Wright-Patterson AFB, Ohio, 45433, Contract No. F33615-73-C-5173. (Grumman Aerospace Corp.)
14. Herman Erbacher, Letter AD C PM 78 195, Grumman Aerospace Corporation, Advanced Composites Group, September 1978.
15. Les Murrin, Letter AD C PM 79 16, Grumman Aerospace Corporation, Advanced Composites Group, January 1979.
16. M. N. Botkin, B. J. Colvin, W. B. Finuf, and R. B. Kollmansberger, F-15 Composite Speedbrake, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio, 45433, AFFDL-TR-75-133, Nov. 1975. (McDonnell Aircraft Co.)
17. Advanced Composites Cost Estimating Model, Air Force Flight Dynamics Laboratory, Wright Patterson AFB, Ohio, 45433, AFFDL-TR-75-3103, August 1976. (Northrop Corp.)

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