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# Composite Structure Repair

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ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT  
(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Report No.716  
COMPOSITE STRUCTURE REPAIR

by

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Paper presented at the 57th Meeting of the Structures and Materials Panel in  
Vimeiro, Portugal on 9-14 October 1983.

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## PREFACE

In preparation for a Specialist Meeting being planned for spring 1986, a pilot paper on the subject "Composite Structure Repair" was provided to "The Repair of Aircraft Structures Involving Composite Materials" Sub-Committee at the 57th meeting of the AGARD Structures and Materials Panel. Mr Larry Kelly, USAF, Air Force Wright Aeronautical Laboratories, presented in the pilot paper a summary of USAF experience in repairing in-service aircraft structural composites. This paper has assisted the panel in defining the context which should be emphasized in the Specialist Meeting and the Sub-Committee is grateful for Mr Kelly's assistance.

KEITH I. COLLIER  
Chairman, Sub-Committee on  
The Repair of Aircraft Structures  
Involving Composite Materials



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Composite Structure Repair  
Larry G. Kelly  
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Abstract:

The technology for advanced composite structure repair is presently in a developing stage. The boundaries and limitations of bolted versus bonded repairs and precured patches versus cocured in place patches and their applicability to various types of hardware has yet to be clearly established. This paper does not discuss step by step repair procedures for specific aircraft components, such as defined in repair technical orders, but rather provides general guidelines for repair concepts and discusses two repair configurations that are generic in nature; an external patch and a near flush repair and the extent to which they have been verified in the U.S. These repairs are applicable to a wide variety of light to moderately bonded (up to 25,000 lb/inch) stiffened and honeycomb sandwich structure sustaining damage over a reasonably large area (up to 100 sq. in.). Also provided are references to documents containing step by step procedures for these repair techniques and identification of organizations in the U.S. actively engaged in advanced composite structure repair.

Introduction:

Major airframe components built of advanced composite materials are presently flying on a number of military production aircraft in the U.S. Use, in the Air Force, began with the F-15 and F-16 aircraft which employ 1.6% and 2.5% advanced composites by structural weight. The Navy's F-18 and AV8B aircraft extended the use to 9.5% and 26% respectively. The Army is presently evaluating a composite rear fuselage for the Black Hawk UH60 helicopter which would extend the amount of composite structure utilized from 17% to 26%. This includes fiberglass, Kevlar and carbon materials. The Army is also developing a composite helicopter prototype under the Advanced Composite Airframe Program (ACAP) that will utilize composites for 75-80% of the airframe by weight.

Until recently, advanced composite parts subject to major damage, were returned to the manufacturer for repair. This situation is rapidly changing for all three services are preparing to maintain aircraft that make extensive use of composite materials. Advanced composites are now being considered, in the U.S. aircraft industry, for all aircraft structure applications where substantial weight savings, stiffness or design efficiency requires tailoring the structure for anisotropic load requirements.

U.S. Advanced Composite Repair Experience:

The service experience with advanced composites has been generally good with the exception of a few parts. Maintenance problems, for the most part, have consisted of edge damage or punctures and dents on composite covered honeycomb. These have been readily repaired by both field and depot level personnel. These repairs have been generally non structural, that is, performed to prevent damage growth, provide aerodynamic smoothness or prevent

moisture intrusion. The bounds for such repairs have been adequately defined by appropriate technical orders. Several military repair centers are rapidly developing the capability to do much more extensive repair and even major composite structure remanufacture if necessary. Some of the more noteworthy facilities in this regard are:

- Naval Air Rework Facility - North Island, San Diego, California
- Naval Air Rework Facility - Cherry Point, North Carolina
- Air Force Logistics Center - Warner Robins AFB, Georgia
- Air Force Logistics Center - Hill AFB, Ogden, Utah
- Air Force Logistics Center - McClellan AFB\*, Sacramento, California
- Army Depot - Corpus Christi, Texas

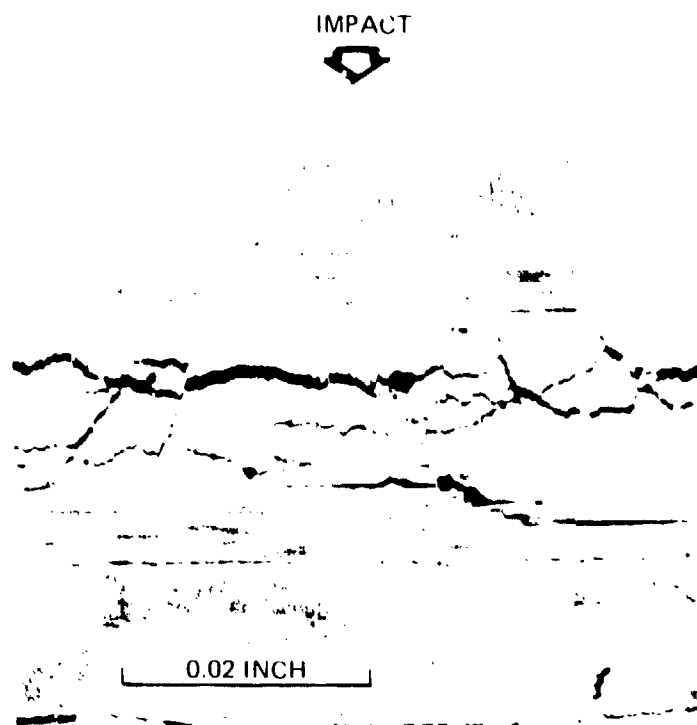
These facilities are being supported by several Research and Development organizations with background experience in advanced composites. The following R&D organizations are actively involved in composite repair technique development:

- Naval Air Development Center - Warminster, Pennsylvania
- Naval Research Labs - Washington, DC
- Air Force Wright Aeronautical Labs - Dayton, Ohio
- Army Applied Technology Lab - Fort Eustis, Virginia

#### Damage Assessment:

General impact damage and specifically ballistic penetration of a composite laminate results in holes in the laminate which are irregular in contour and generally jagged in appearance. Delaminations, void areas and ruptured filament bonds may occur anywhere throughout the thickness, but generally to a larger extent on the opposite side of the impacted face or exit side of the projectile path. In some cases, impacts which cause very little damage on the surface can cause internal cracking and delamination. These interlaminar defects can be readily detected with ultrasonic equipment and it is a good rule that any damage which is visible on the surface should be further evaluated for internal damage. Examples of extensive internal damage where surface damage is minor are shown in Figures 1 and 2.

\*This facility was recently designated by the Air Force to be its lead center for establishing composite repair training requirements for ALC engineers and maintenance personnel. The center will develop composite repair techniques including training and equipment needs, be a focal point for overall composite repair technology and aid the other ALCs in implementing composite repairs.



8 PLIES GR/EP ( $\pm 45/0/90$ )<sub>s</sub>

BLUNT IMPACTOR AT CENTER OF 5 INCH SQUARE AREA  
 TOTAL ABSORBED ENERGY = 1.24 FT-LB  
 (INCIPIENT DAMAGE INDICATED AT 0.82 FT-LB)  
 DAMAGE NOT VISIBLE ON IMPACTED SURFACE  
 SLIGHT MATRIX CRACK ON BACK FACE

Figure 1. Photomicrograph of Impact Damage Laminate

Face Sheet: 4 Ply HMF 133  
 Woven GR/EP

Surface Indentation = 0.021 inch  
 (No cracks or broken fibers)

FM 300 Adhesive

Impacted by 5/8 inch diameter  
 Spherical Steel Impactor

Impact Energy = 1.78 Ft-Lb.

Impact



Core: 6.1 PCF A1 Honeycomb

Figure 2. Photomicrograph of Impact Damaged Honeycomb Panel

The primary field and depot inspection methods being utilized in the U.S. for composite structure are through transmission and resonance ultrasonics and radiography. Radiography inspection is used to detect broken bondlines (core splice and core to closeout members) and to detect the presence of water in the core cells. It can also be used to detect porous or excessively thick bondlines and deformed core.

Ultrasonic equipment is the most widely used and is generally employed with a set of standards for set up and defect comparison. Figure 3 compares size of visible damage to area of internal delaminations as determined by ultrasonics. This data is from Reference 1 and is for a wide range of carbon panel types; some with buffer strips and stitching to contain delayed and superquick fuzed 23 mm high explosive projectile damage. The original data is from McDonnell Aircraft but I have included data from Boeing, Northrop and Air Force reports (dots and bars). This data includes impacts of fragments (1/4, 3/8, 1/2 inch) and projectiles (12.7, 14.5, 23 mm) with angles of obliquity up to 60 degrees.

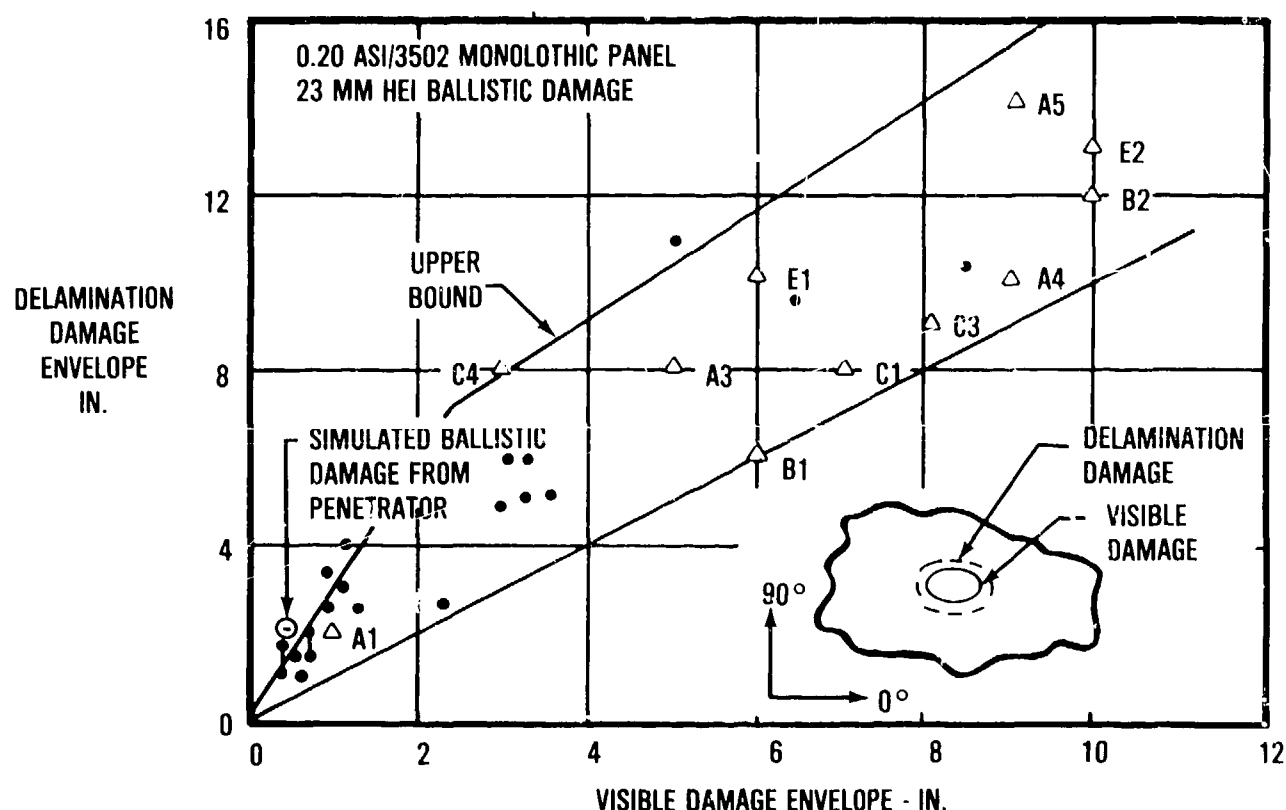


Figure 3. Ballistic Damage to Carbon Epoxy Panels

Some test results, Reference 2, indicate that for a given panel width and laminate orientation various through-the-thickness crack geometries having the same crack width, as shown in Figure 4, failed at essentially the same tension load. Thus assuming damage to consist of a through-the-thickness defect equal in width to the maximum damage dimension (as determined by ultrasonics) perpendicular to the primary load path, linear elastic fracture

mechanics can be utilized to obtain an estimate of the strength lost. This approach of modeling damage effective strain concentrations as that of an equivalent open round hole can sometimes be unconservative but a useful technique to obtain a "ball park" estimate of how much strength has been lost.

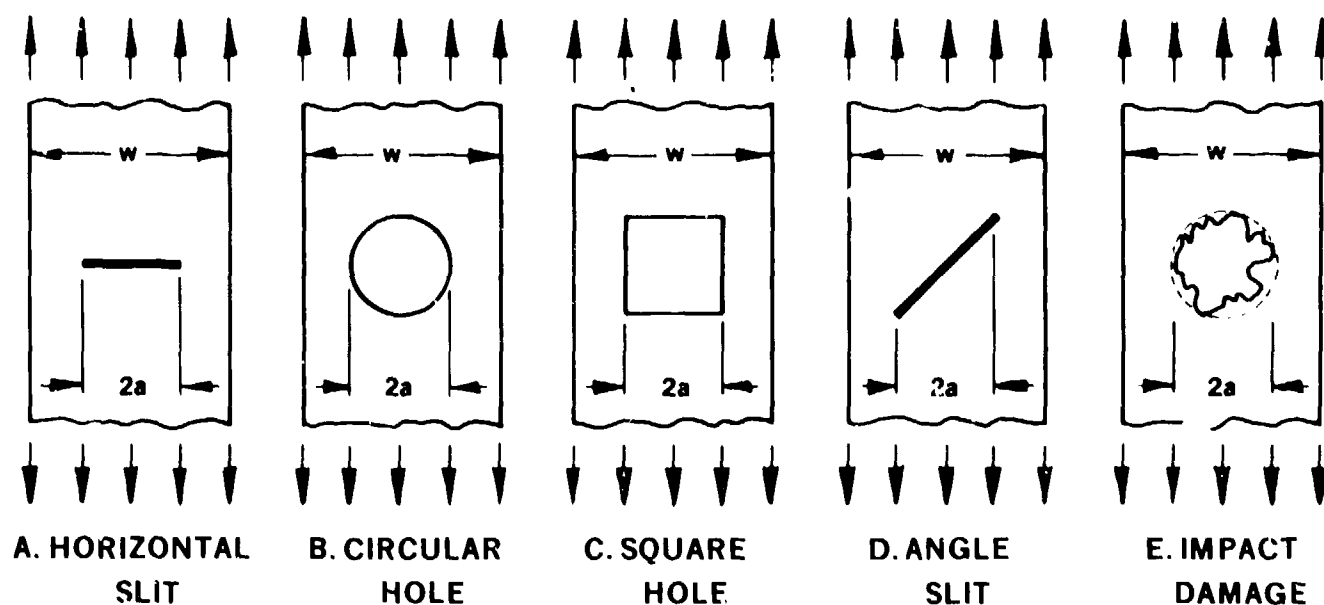


Figure 4. Examples of Through-Thickness Defects Having Same Tension Failure Loads

Figure 5, from Reference 3, shows that there is substantial strength loss in carbon composites with relatively small holes.

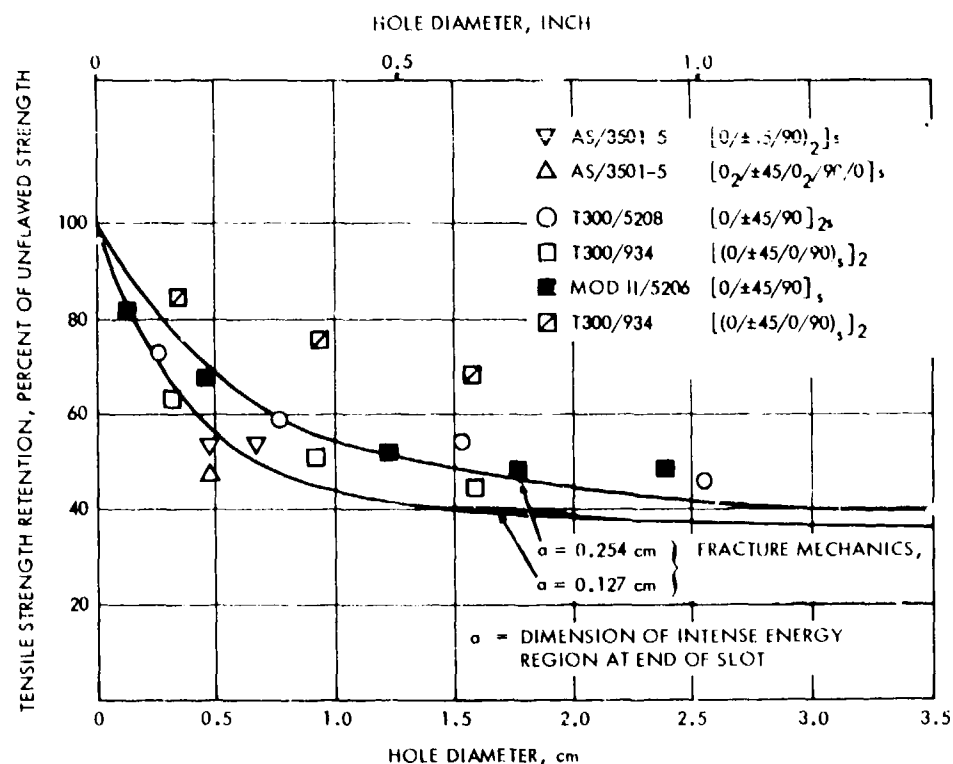


Figure 5. Tensile Strength Retention of Laminates with a Hole

Current design practice in the U.S. is to limit ultimate strain allowables in carbon composite structures to approximately 3500-4500 $\mu$  in/in. This allows for stress concentrations due to bolt holes or notches and provides for some accommodation of strain concentrations due to defects or damage. The figure 5 data does, however, point out the need for good quality repairs with substantial load carrying capability especially for structures designed with higher strain allowables.

#### Bonded Repairs:

Two types of bonded repairs are discussed below: 1) a nearly flush repair for which a scarf joint surface is machined in the parent laminate and replacement plies with adhesive are cocured into place; 2) an external patch which is precured and subsequently bonded over the damaged area. These repairs can be used for on aircraft or off aircraft repairs, for repairs accessible only from one side for either flush or external patches and for both monolithic or sandwich construction. The information provided is not intended to be a step by step guide for repair patch installation such as found in References 4 and 5, but rather a discussion of standardized repair procedures that have been verified, and general engineering guidance for the designer of the repair patch.

After assessing the damage and before deciding upon a repair, the question of the parent laminate moisture condition becomes important. Moisture absorbed in the laminate and/or entrapped moisture in honeycomb can be very detrimental to the integrity of bonded repairs. Examination of cured carbon epoxy patches bonded to substrates containing moisture, similar to long term service experience in a high moisture environment, showed a porous bond line. See figure 6:



Figure 6. Porosity in AF-147 Bondline on 50-Ply Wet Laminate

This absorbed moisture has had detrimental effects on repairs in the following four ways:

- 1) Local delamination or blistering in parent laminates
- 2) Reduced strength of the repair and repair bond line resulting from porosity.
- 3) Expanding moisture in honeycomb cells has created sufficient pressure to separate the skin from the core.
- 4) Reduced effectiveness of ultrasonic inspection due to strong signal attenuation making it difficult to verify bond line integrity.

Prebond drying (a minimum of 48 hours at 170°F-200°F), slow heat up rates, reduced cure temperatures and selection of adhesives less sensitive to moisture can minimize or eliminate the above problems. The 250°F curing adhesives, as a group, are more sensitive to prebond moisture at higher temperatures (above 150°F) than 350°F curing adhesives. Drying the parent laminate to an average moisture content of less than .5 percent is recommended. This can be very time consuming taking over 24 hours for a 16 ply laminate, as shown in figure 7.

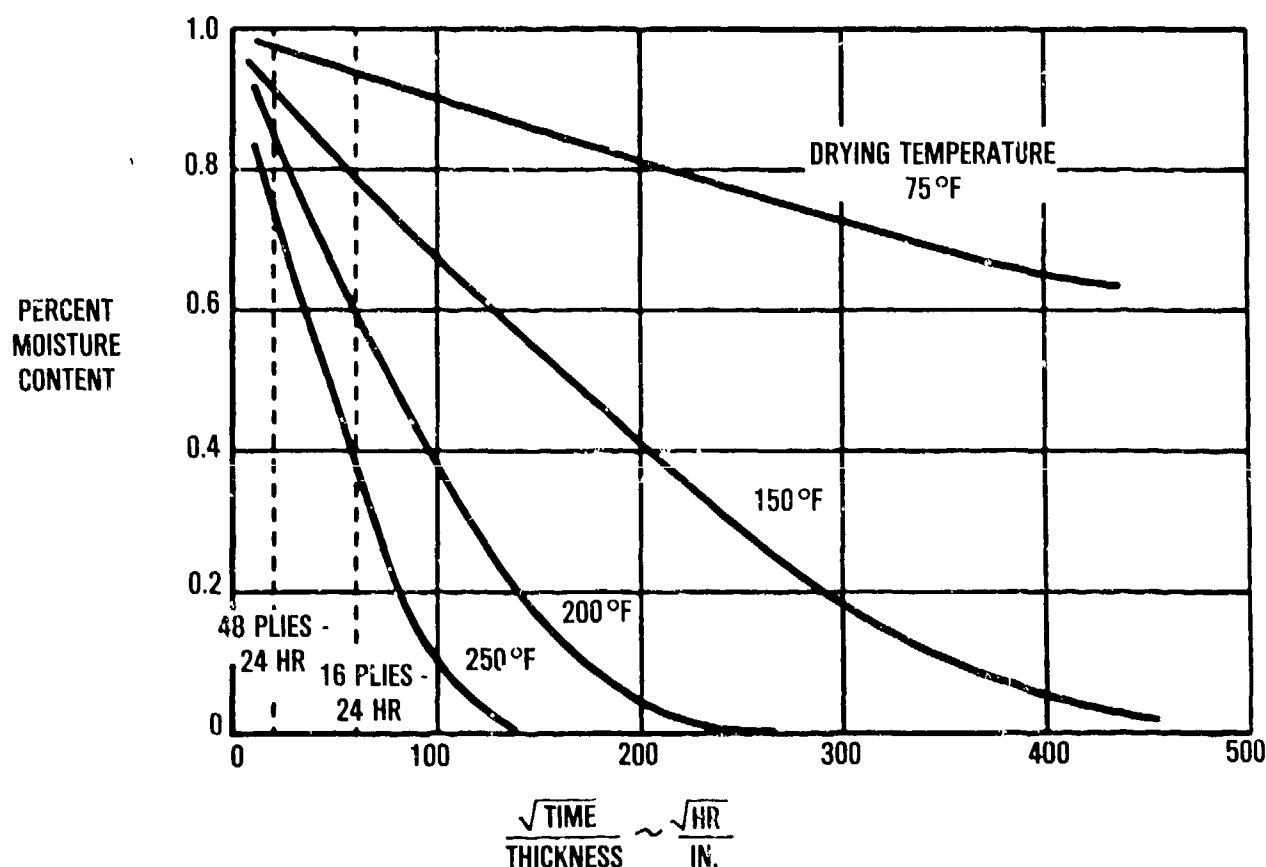


Figure 7. Drying Time for Carbon Epoxy Laminates

External bonded doublers are the simplest to apply. Their load carrying capability is, however, somewhat limited for no matter how well the edge of the patch is tapered the edge of the parent laminate, at the hole, is a point of high shear and peel stress concentration. Since the interlaminar tensile strength of carbon epoxy is less than the peel strength of typical structural

adhesives, the effect of this failure mode is to restrict the thickness of composites which can be bonded efficiently using standard lap joints. Reference 6 shows that peeling can be minimized by small fasteners at about one inch spacing around the hole and a router cut about .017 inch deep filled with adhesive and a ply of fiberglass prepreg. This concept shown in Figure 8, on a 16 ply  $[(+45/0/90)_2]$  laminate using 1/8" blind rivets raised the joint efficiency from 52% to 78%.


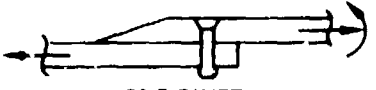

FM-400 BONDED JOINTS (WIDTH = 1.00 INCH)	FAILURE MODE	JOINT EFF
 CONTROL	PEEL AND SHEAR FAILURE	0.52
 ONE RIVET	SHEAR AND PATCH NET TENSION FAILURE	0.73
 UNDERCUT & ONE RIVET	SHEAR AND RIVET HEAD PULL-THRU FAILURE	0.78

Figure 8. External Patch Concepts

Applying this concept to 2" diameter holes in four-point-load sandwich beams obtained the results shown in Table 1. Figure 9 shows the details of the 22 ply patch design. Comparing the resulting failure loads with the parent allowables quite satisfactory results were obtained for low to intermediate load levels (6-7,000 lb/inch).

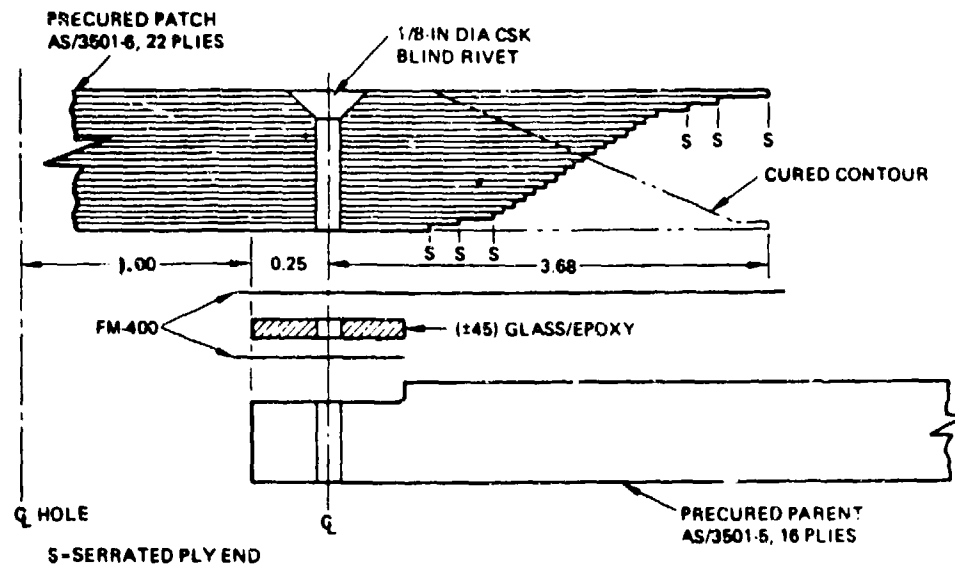


Figure 9. External Patch Repair

### External Patch Repair Test Results (Four Point Load Beam Specimens)

TEST SERIES	TEST TEMP (°F)	JOINT MOISTURE COND	FATIGUE		LOAD SENSE	ULTIMATE LOAD (LB/IN)	FAILURE MODE (b)	PARENT ALLOWABLE (LB/IN)	JOINT EFF
			MAX LOAD (LB/IN)	LIFE- TIME (a)					
I	-65	Dry	None		Tens	6230 6350 6390 Avg. 6360	FTI PTI, B PTI, B	6210	1.02
II	RT	Dry	None		Tens	6610 6300 Avg. 6406	PTI PTI	6860	0.93
III	265	Dry	None		Tens	5820 6140 5960 Avg. 5970	AI AI AI	5750	1.04
IV	RT	Dry	5810 5810 { 5810 2820 2820	0.10 0.12 0.17 } 1.83 2.0	Tens  Tens	*** *** 6440 6500 Avg. 6470	PTI PTI PTI PTI	6860	0.94
V	-65	Dry	None		Comp	8050 7430 7630 Avg. 7670	PC PC PC	9740	0.79
VI	RT	Dry	None		Comp	7200 6930 6970 Avg. 7030	PC PC PC	7600	0.92
VII	265	Dry	None		Comp	4890 4990 5440 Avg. 5040	PC PC PC		
VIII	RT	Dry	2820 2820	2.0 2.0	Comp	7270 7070 Avg. 7170	PC PC	7600	0.94

(a) Tension spectrum fatigue, F-5E wing lower skin root.

(b) Failure modes:

PC = Parent laminate compression failure near edge of patch.

PTI = Parent laminate tension and interlaminar shear failure at edge of patch.

AI = Adhesive and/or interlaminar shear failure.

B = Blister repair left some unbonded laminate involved in failure area

\*\*\* Failed in fatigue loading due to unintended high load application.

Precured bonded composite doublers, discussed previously, metal sheets and plates and metal foils of 8, 12 and 16 mil thickness have all been utilized satisfactorily as patch materials for low load levels and relatively flat surfaces. For curved or irregular surfaces and intermediate to high load levels the most versatile repair concept is the cured in place flush scarf repair. Accomplishing the scarf may appear difficult at first but was found to be relatively easy with simple portable tools. As a general rule when making a large area repair a flush scarfed repair is preferred since a significantly higher percent of the strength of the parent laminate can be restored. This is especially true for compressive loading where eccentricity of the patch can increase bending forces. Practical size limitations will probably restrict this repair approach to laminate thicknesses less than 1/2 inch because of the amount of material that has to be removed to achieve the required taper for proper scarf angle ( $L = 18$  to 40 times the thickness). Figure 10 illustrates a typical scarf joint using a 16 ply laminate to be repaired.

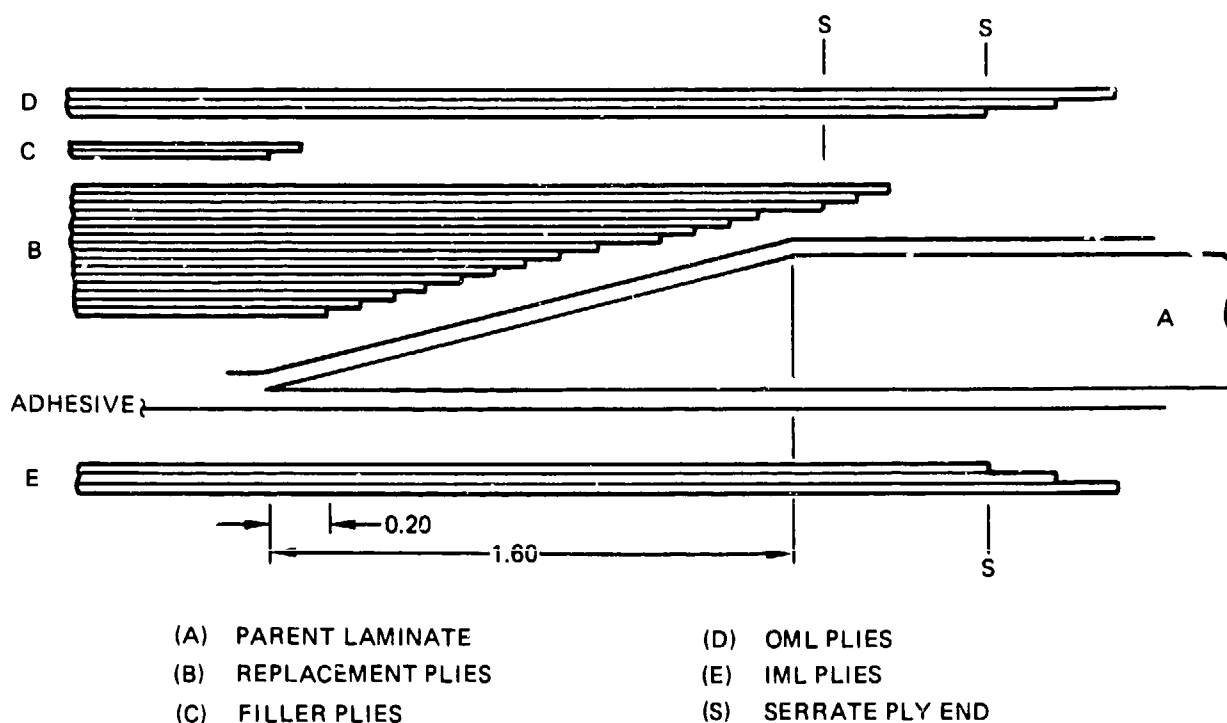


Figure 10. Basic Cured 16-Ply Scarf Joint

This basic scarf joint employs an 18:1 taper ratio since a .10 inch step is 18 times the nominal ply thickness of (.0055 inch). The scarfing is accomplished quite readily with the use of a portable router to cut .10 inch concentric steps, each successively larger, followed by a portable power driven sander to provide a finished scarf. A good scarf patch design practice is to extend the outermost plies over the ends of all the other plies and to serrate these plies to minimize ply end peeling. This can easily be done to the edge normal to the fiber direction with a pair of standard 1/8 inch V-notch pinking shears. It is also good practice to avoid placing unidirectional material as the outer most ply. An outside cover of woven material or for balanced

laminate layups consisting of  $0^\circ$ ,  $90^\circ$ , and  $+45^\circ$  layers, the high strain low modulus ( $+45^\circ$ ) layers should go on the outside. This makes surface defects such as cuts, scratches and abrasions less strength critical. One other point to remember is that laminates cured with vacuum pressure only, tend toward void contents of about 5 percent, as compared with less than 1 percent voids for laminates cured at 100 psi in an autoclave. The higher void content reduces strength properties by approximately 15% for the vacuum cured material and this strength reduction should be considered in developing the repair.

Repairs up to 100 inch sq. in 16 and 24 ply laminates have restored 80-100 percent of the parent laminate allowable utilizing the techniques described above. This has been verified through a series of repair joint coupons, sandwich beams, flat panels and box beam tests. First 1" wide tension coupons and compression sandwich beam specimens of the scarf design shown in Figure 11 were tested with the results shown in Table 2.

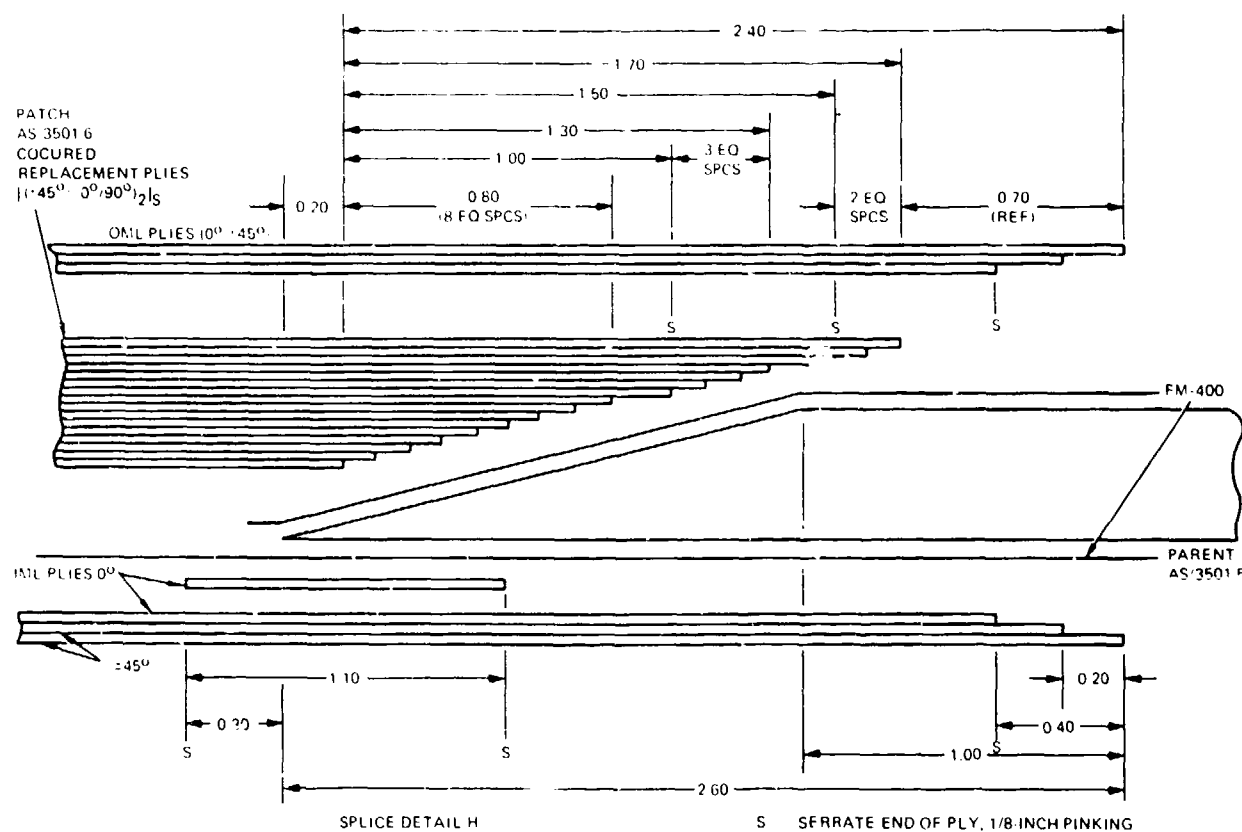


Figure 11. Flush Scarf Repair

The test results shown in Table 2 are for specimens consisting of a parent laminate of AS/3501-5  $[(0/+45/90)_2]_S$  that was scarfed over a length of 1.6 inches and a patch laminate of AS/3501-6 plies, per Figure 11, that was cocured and bonded to the scarf. Comparing the failure loads for these joints to that of the parent laminate allowable, quite satisfactory joint load transfer efficiencies were achieved.

Table 2  
Single Scarf Test Results

TEST TEMP (°F)	PARENT LAMINATE	(b) FATIGUE MAX LOAD (LB/IN)	LOAD SENSE	(c) AVERAGE FAILURE LOAD (LB/IN)	PARENT ALLOWABLE (LB/IN)	JOINT EFF
-65	$[(0/\pm 5/90)_2]_s$ ↓	None	Tension (Coupon)	4970	6210	0.80
RT		None		5638	6860	0.83
RT		None		6665	6860	0.97
265		None		6333	5750	1.10
RT		1550		6487	6860	0.95
RT		2820		6123	6860	0.87
-65	$[(+45/0/90)_2]_s$ ↓	None	Compr. (Beam)	8471	9740	0.87
RT		None		7213	7600	0.95
265		None		6230	4710	1.32
RT		1550		7636	7600	1.00
RT		2820		7830	7600	1.03
-65	$[(+45/0/90)_2]_s$ ↓         *WET ↓	None	Tension (Coupon)	6706	6210	1.08
RT		None		6890	6860	1.00
265		None		5057	5750	0.88
RT		1550		6875	6860	1.00
RT		2820		7136	6860	1.04
-65		None	Compr. (Beam)	8076	9740	0.83
RT		None		5988	7600	0.79
265		None		4946	4710	1.05
RT		1550		6830	7600	0.90
RT		2820		7165	7600	0.94

(a) Splice details shown in Figure 11.

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

(c) Average of three replicates.

\*WET - The repair assembly was moisture conditioned at 95% relative humidity and 140°F for 30 days before testing.

Having established the capability of composite scarfed joints to satisfactorily transfer the required loads, the application of this repair concept (Figure 11) and that of the precured external patch (Figure 9) was applied to the repair of 4 inch diameter holes in 12 inch by 48 inch panels (Figure 12).

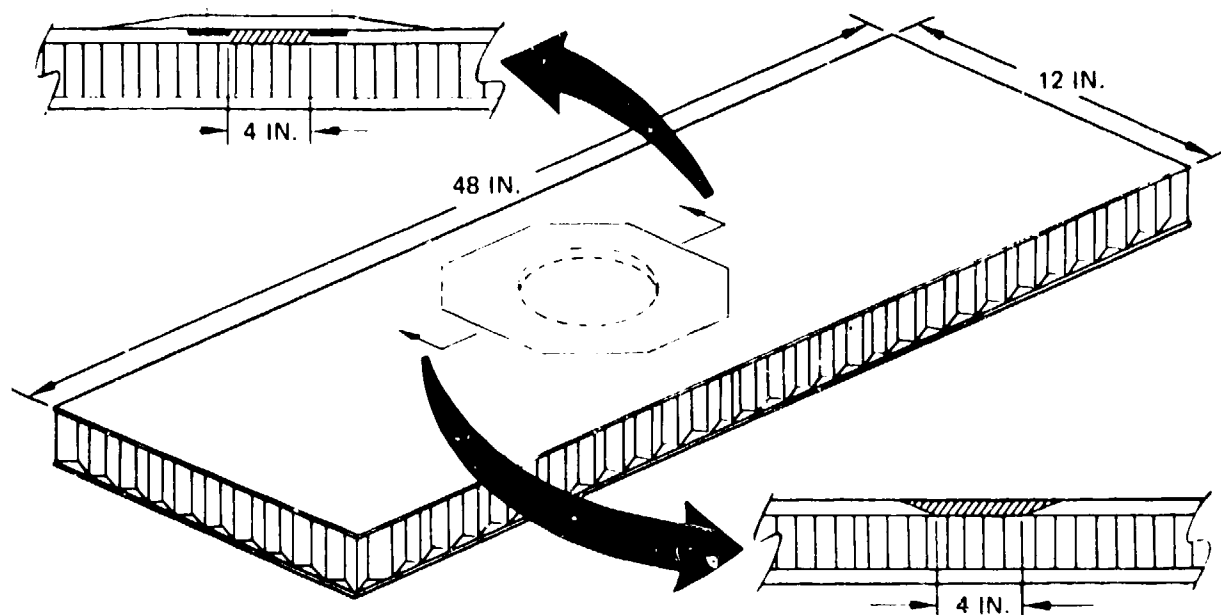


Figure 12. Intermediate Size Panel Repairs

The results of these panel tests are shown in Tables 3 and 4. All 14 panels were tested as four point beams with either a constant applied tension or compression moment in the repair section. The better load transfer capability of the flush scarf repair is evident even at these low load levels. The precured external patch performance in compression was unsatisfactory in terms of load restored, only 66-69% of the parent allowable.

The final demonstration of the repair concept developed was accomplished through the use of a 17 foot long, 19 inch wide, 7 inch deep box beam to which four point bending and torsion loads can be applied to the tension cover of a five foot test section, Figure 13. The five test panels and repair techniques utilized are outlined below:

- Panel 1. Honeycomb Sandwich Panel (8 ply laminates on .5 inch thick aluminum core)  
 Damage: 6" x 12" oval hole completely through after clean up  
 Repair: 13 ply bonded cocured scarf patch to both facesheets (36:1 taper) + core plug
- Panel 2. Honeycomb Sandwich Panel (8 ply laminates on .5 inch thick aluminum core)  
 Damage: 6" x 12" oval hole after clean up on front face and a 1" x 6" hole on the back face  
 Repair: 5 ply precured blind side patch + 13 ply bonded cocured scarf patch for inner and outer faces + core plug

PANEL NO.	STRUCTURAL DEPTH (IN.)	REPAIR MOISTURE CONDITION	LOAD SENSE	ULTIMATE LOAD, 2P (LB)	FAILURE MODE (a)	ULTIMATE FACE LOAD (LBS/IN.)	PARENT ALLOWABLE (LBS/IN.)	JOINT EFF.	COUPON OR BEAM JOINT EFF.
2101	1.81	DRY	TENSION	22,000	PT	7600			
2102	1.81	DRY	TENSION	22,000	PT	7600			
2104 *	1.81	DRY	TENSION	21,100	PT	7290			
						AVG. 7500	6860	1 09	0 97
2103	1.79	DRY	COMPRESSION	20,100	RCP	7020	7600	0 92	0 95

(a) FAILURE MODES.  
PT = PARENT LAMINATE TENSION,  
RCP = REPAIR COMPRESSION AND PEEL

SCARF MACHINED ON A MILL, ALL OTHERS PREPARED WITH ROUTER AND SANDER.

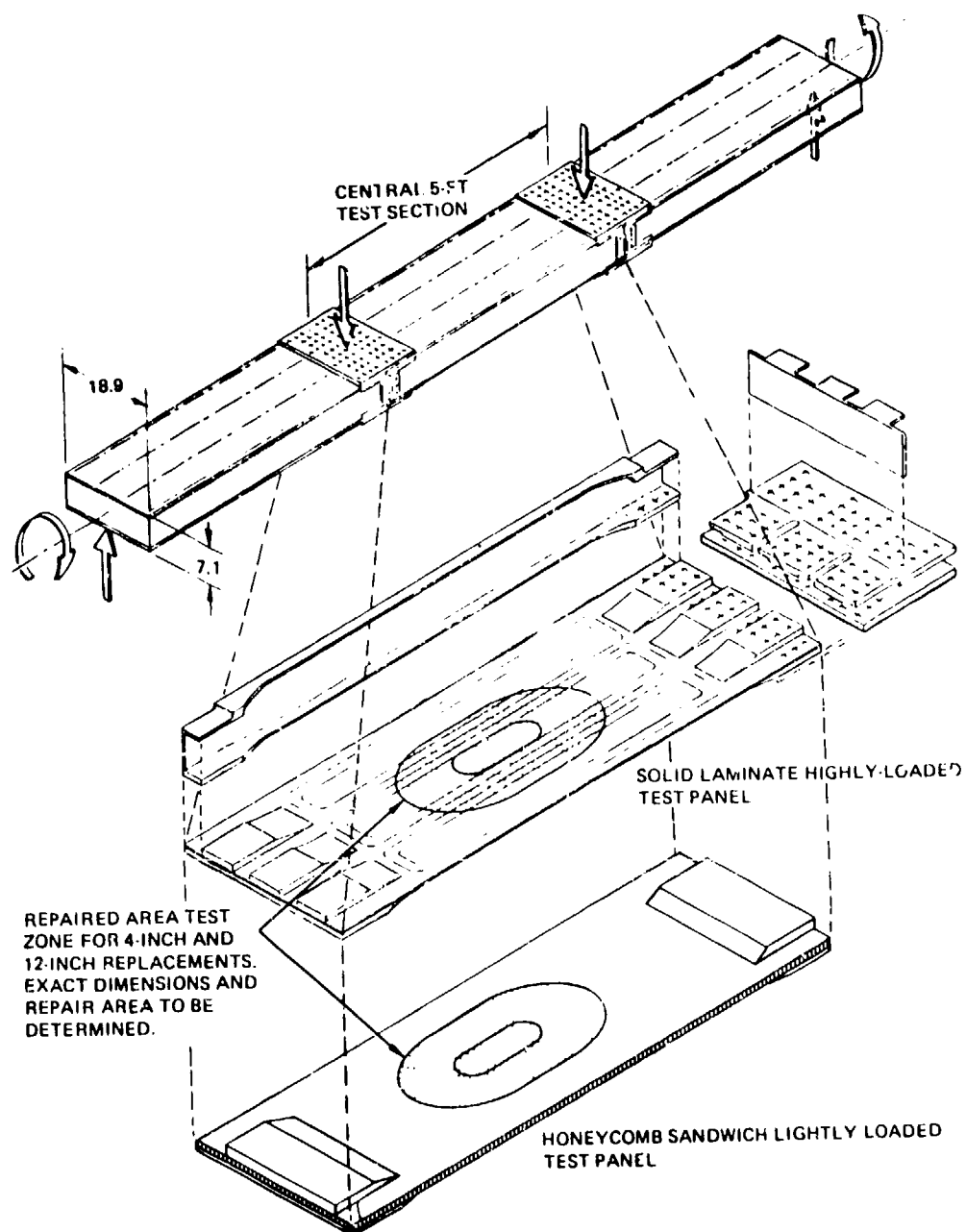


Figure 13. Large Scale Demonstration Panel  
Box Beam Test Fixture

Panel 3. 50 Ply Laminate

Damage: 8" x 12" oval 21 plies deep after clean up

Repair: 24 ply bonded cocured scarf patch (36:1 taper) at ends  
(18:1 taper) at sidesPanel 4. 50 Ply Laminate

Damage: 4" diameter hole completely through

Repair: 61 ply cocured double scarf patch (36:1 taper)

Panel 5. 64 Ply Boron/Carbon Epoxy Hybrid Laminate

Damage: 9" diameter 24 ply cut out after clean up

Repair: 77 ply bonded cocured scarf patch (55:1 taper) at ends  
(18:1 taper) at sides

Test results for these five panels are shown in Figure 14 and summarized below. More details on these test results can be found in Reference 6.

Panel 1. Failure was remote from the patch area at 139% of the parent ultimate tension allowable at a failure load of 2390 lbs/inch

Panel 2. Failure was through the repaired area at 122% of the parent ultimate tension allowable at a failure load of 2095 lbs/inch

Panel 3. Failure was remote from the patch area at 155% of the parent ultimate tension allowable at a failure load of 14,200 lbs/inch

Panel 4. Failure was through the repaired area at 155% of the parent ultimate tension allowable at a failure load of 14,200 lbs/inch

Panel 5. Failure was remote from the repaired area at 151% of the parent ultimate tension allowable at a failure load of 19,300 lbs/inch

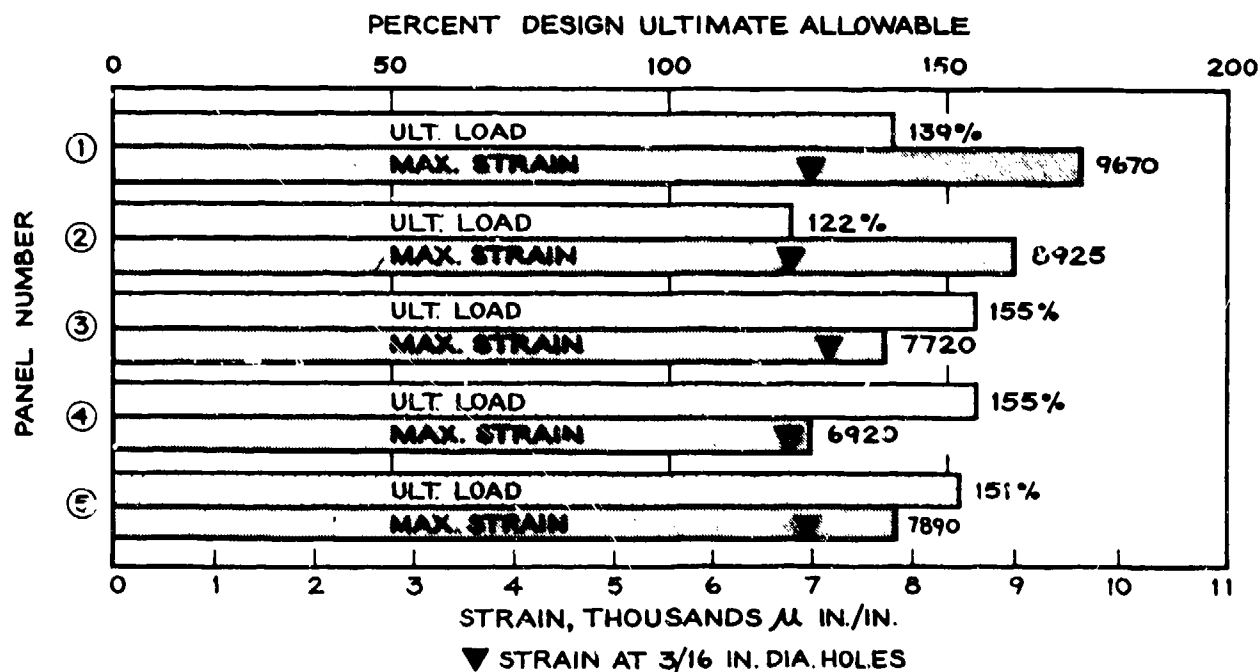


Figure 14. Demonstration Panel Results

Summary:

This paper has addressed solely bonded repairs and concentrated on two concepts; a precured external bonded doubler and a flush scarf cocured patch. The test results presented validate the flush scarf repair concept as a viable repair approach and in fact, this repair technique is presently being utilized at Air Force Logistic Centers.

Several programs have also been conducted on bolted repairs and this could be the subject of a future paper. In addition, current programs not yet complete, are addressing repairs of thicker more highly loaded structure (up to 80 ply laminates) subjected to multiple impacts. A combination bond-rivet approach, is also, being evaluated for repairing delaminations and providing damage confinement or a fail safe mechanical load path for high loaded bonded structure (Reference 7). Considerable work has been accomplished on bonded aluminum honeycomb sandwich structure repair (Reference 8) and bonded skin stringer frame construction (Reference 9). Finally several organizations have shown that composite patches applied to cracked metallic structure are very effective in extending fatigue life (Reference 10). Thus there are several alternative repair concepts any one or all of which can be addressed by this AGARD panel, so some boundaries will have to be decided on as far as the scope of repair activities appropriate to the panel.

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