NASA Technical Memorandum 84505

A Repair Technology Program at NASA on Composite Materials

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SUMMARY

The repair of structures fabricated from composite materials is a many-faceted problem. The research includes identifying defective areas, determining whether the damage is detrimental to structural performance, developing efficient repair procedures, and performing tests to demonstrate that the repair will allow the component to be put back in service for the remainder of the design life. This paper discusses the repair technology program at the Langley Research Center for both graphite/epoxy and graphite/polyimide composite structures. Research at Langley, contracts with aerospace companies, and grants with universities are being used to investigate the repair problem. The compressive behavior of laminates, hat-stiffened panels, and honeycomb sandwich panels that have internal delaminations is presented. The tensile and compressive strengths of repaired laminates for several basic repair processes are compared and are shown to approach the strength of the undamaged laminate. Other repair configurations currently under investigation are illustrated, and plans for future work in the repair technology program are presented.

INTRODUCTION

The repair of structures fabricated from composite materials is a many-faceted problem. The research includes identifying defective areas, determining whether the damage is detrimental to structural performance, developing efficient repair procedures, and performing tests to demonstrate that the repair will allow the component to be put back into service for the remainder of the design life. Research at Langley, contracts with aerospace companies, and grants with universities are being used to investigate the repair problem. The repair program focuses both on graphite/epoxy (Gr/E) composites for commercial transport applications and on graphite/polyimide (Gr/PI) composites for higher temperature aerospace applications.

The need for composite-material repair has been demonstrated in the NASA flight-service program. In the early 1970's NASA recognized the need to increase confidence in the long-term durability of advanced composites that would encourage aircraft manufacturers and operators to make production commitments for the use of composite structures on commercial aircraft. Thus, NASA initiated a systematic program for the design, fabrication, testing, and flight-service evaluation of numerous composite components to help increase the necessary confidence. Major emphasis was placed on the evaluation of advanced composites on commercial transport aircraft because of their high utilization rates, exposure to worldwide environmental conditions, and systematic maintenance procedures. Some components that are being evaluated on commercial transport aircraft are shown in figure 1, and a summary of NASA's composite-structures flight service through March 1982 is given in table I. Additional details of NASA's flight-service program are given in reference 1. In 10 years, a total of 155 composite components have been put into flight service and have amassed over 2-million flight hours. Inspection and maintenance results are given in table II. Flight-service components on commercial aircraft are inspected every 12 or 13 months, and components on military aircraft are inspected at 2- or 6-month intervals. All components are inspected visually and some are also inspected with ultrasonics. The use of advanced composites on more complicated structural components may require other inspection methods such as X-ray or eddy current. (See refs. 2 and 3.) Data from table II indicate that some composite components from all the flight-service programs except the Lockheed C-130 center-wing box have experienced either damage or deterioration in the flight-service environment. Figure 2 illustrates some of the types of damage that have occurred on some of the Boeing 737 spoilers during 9 years of flight service. In one case, a cable cut through the trailing edge, and, in another case, sufficient clearance was not provided for an actuator-arm end fitting so that it punctured the upper-surface skin. Other types of damage include foreign-object impact. In all cases where repairs were necessary, the spoiler was removed and returned to the manufacturer for repair.

The increased usage of advanced composite structures on commercial aircraft is anticipated and will require the development of generic repair techniques and processes for various types of structures to enable the airline operators to maintain the components. The purpose of this paper is to provide an overview of the research related to repair technology of advanced composite structures sponsored by NASA. Contained herein are selected results from the repair technology program that indicate that efficient repair techniques have been developed and demonstrated on coupon and larger subelement specimens applicable to secondary Gr/E structures for commercial transport aircraft. The technology base developed for repair of Gr/E structures is being applied to repair technique development for Gr/PI structures for higher temperature applications. Future work in the repair of Gr/E will address large-area repair and evaluation on a large full-size composite component. The long-term durability of repairs on Gr/E will be evaluated in a 10-year outdoor-exposure program at the Langley Research Center.

SYMBOLS

a	delamination width
w	panel width
σ	compression strength
σ_f	flawed compression strength

Abbreviations:

Gr/E	graphite/epoxy
Gr/PI	graphite/polyimide
RT	room temperature

EFFECT OF DEFECTS

An important consideration is whether a detected flaw is detrimental to the safe use of the structure and whether a repair can be made. This problem is addressed through the testing of thin laminates, stiffened elements, and honeycomb panels. Manufacturing and service-damage-type flaws are being investigated for Gr/E and Gr/PI structural elements. Laminates of 6, 8, 12, and 16 plies are being used to determine the effect on compressive strength of the following types of flaws: interlayer delaminations, surface cracks, delaminations between Gr/PI and metal inserts, and impact damage. These four flaw types were also used in the study to determine their effect on performance of stiffened elements and honeycomb-core panels representative of current and near-term aerospace applications. Figure 3 shows a typical set of data for a laminate, a hat-stiffened panel, and a honeycomb-core panel that has simulated delaminations and is subjected to compression. The delaminations were either 161, 645, or 2580 mm^2 in area and were made by placing one piece of 0.976-mm heat-treated Kapton[1] film on top of another during the fabrication process. All three specimens illustrated in the figure were 0.15 m wide by 0.30 m long. The data shown for the laminate are for the graphite/epoxy T300/5208, and data shown for the hat-stiffened and honeycomb-core panels are for the graphite/polyimide Celanese Celion 6000/PMR-15. The compression strength, normalized as a percent of unflawed strength, is plotted as a function of the ratio of delamination width to panel width. For a 5-cm delamination in the laminate, a 30-percent reduction in the compression strength was observed, whereas for the same size of delamination in the hat-stiffened and honeycomb-core panels, 40-percent and 60-percent reductions in the compression strength were observed, respectively.

The data shown are for one part of an ongoing effort to determine the effects of various types of damage on composite structural performance. The Langley Structures and Dynamics Division is investigating the compressive behavior of composite panels subjected to local low-velocity impact. (See refs. 4 and 5.) Test results suggest that lightly loaded panels, which may be designed by stiffness requirements, may achieve their design strength in the presence of impact damage. Heavily loaded, strength-critical panels, however, experience significant strength reductions because of impact damage that is not visually detectable. The Fatigue and Fracture Branch of the Langley Materials Division is investigating the fracture toughness of composites and methods for improving damage tolerance. (See refs. 6, 7, and 8.) The fatigue of notched composite laminates (refs. 9 and 10) and the delamination growth in composites (refs. 11, 12, and 13) are also being studied.

REPAIR DEVELOPMENTS

Repair Concepts

After deciding that a detected flaw is detrimental to the use of the structural component, repair procedures must be developed and validated that are adaptable to current maintenance operations and that meet the basic criteria for restoring design strength and service life to the repaired component. Figure 4 gives results for three repair procedures being studied at the University of Delaware under NASA grant NSG-1304, "Damage Repair Technology in Composite Materials." (See ref. 14.) Shown at the top of the figure is a cross section of a scarf repair to a damaged laminate. The damaged area is first cleaned out by a machining operation. A machined precured patch is then adhesively bonded in the cleaned-out area and a doubler of the same material may or may not be added. The data shown are for a graphite/epoxy AS/3501-6 laminate having a ply orientation of $(0_2/\pm45/90/\pm45/0_2)_s$. The first bar shows a 50-percent reduction in tensile strength for a specimen that has a 1.3-cm slot. The next bar shows that a plain-scarf repair for a severed laminate restores 65 percent of the tensile strength. The next bar shows that the addition of a uniform doubler does not increase the tensile capability over the plain scarf. This is due to the existence of a large peeling stress at the end of the doubler which initiates failure. However, if the doubler is tapered as indicated in the last bar, then 90 per-

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[1]Kapton: Registered trademark of E. I. du Pont de Nemours & Co., Inc.

cent of the laminate strength can be developed. The data shown are for a scarf angle of 6°; however, a scarf angle of 3° has been shown to be more efficient but much more difficult to machine.

Under contract NAS1-15269, "Development, Demonstration, and Verification of Repair Techniques and Processes for Graphite/Epoxy Structures for Commercial Transport Aircraft," the Lockheed-California Company is developing and evaluating repair procedures for Gr/E composites. Figure 5 shows schematically some of the repair concepts evaluated. Two laminate configurations were used as the parent laminate: a 16-ply laminate, representative of current lightly loaded composite applications; and a 50-ply laminate, representative of highly loaded applications such as a wing cover. The significant variables evaluated in the program, illustrated in figure 5, include flush and external protruding repairs, precured bonded Gr/E patches, cure-in-place Gr/E patches, and a combination of cure-in-place and precured bonded repair. The test specimen used for the repair evaluation was a sandwich-beam specimen which is also shown schematically in figure 5. Damage was simulated as a cut across the laminate width which was rejoined by the repair. (See fig. 5.) All repairs were performed after the parent laminate had been conditioned to a 1-percent moisture content to simulate in-service conditions. In all cases, repairs were made with a 450 K structural adhesive and unaugmented vacuum-cure pressure. Specimens were tested from 218 K to 355 K in static tension and compression. Some fatigue specimens were also included. Figure 6 lists the repair efficiencies obtained for the concepts presented in figure 5 for 50-ply specimens tested at 355 K. The data indicate that the most effective repair concept was the flush, precured Gr/E repair, followed closely by the flush, cure-in-place Gr/E repair. The results indicate that unaugmented vacuum-cure pressure can provide satisfactory structural bonded and cure-in-place Gr/E repairs for laminates representative of highly loaded structures. Additional results from this investigation are given in reference 15.

The bonding of Gr/PI materials is not as well established as the bonding of Gr/E. Therefore, bonded repair techniques for Gr/PI structures are being evaluated through flexure and short-beam-shear tests on Celion 6000/PMR-15 and Celion 6000/LARC-160. The five bonding techniques being evaluated are: (1) cure in place, (2) cure in place with LARC-13 amide/imide (AI) adhesive, (3) secondary bonding with 1 ply of prepreg, (4) secondary bonding with LARC-13AI adhesive, and (5) secondary bonding with LARC-13AI adhesive on 1 ply of Type 104 E-glass carrier cloth. For the cure-in-place techniques, 5 plies of prepreg were cured onto 5-ply precured panels for preparation of the flexure specimens; and 10 plies of prepreg were cured onto 10-ply precured panels for preparation of the short-beam-shear specimens. For the secondary bonding techniques, two 5-ply precured panels were bonded together for preparation of the flexure specimens, and two 10-ply precured panels were bonded together for preparation of the short-beam-shear specimens. In all cases, the bonded panels were 25 cm square. Specimens were tested from control panels and panels bonded with the five techniques. Based on preliminary analysis of the test results, the cure-in-place and the secondary bonding with 1 ply of prepreg procedures offer the most promise for making repairs on both material systems. A typical set of results is presented in figure 7. The data shown are for unidirectional laminates. The short-beam-shear strength indicated for the control panel is the average of 10 tests, and for the cure-in-place bonding it is the average of 5 tests at each of the 5 test temperatures. Superior shear strengths are obtained for the cure-in-place panel at all test temperatures except 116 K. This trend was also observed for the flexure tests. Based on these results, cure-in-place repairs have been made on Celion 6000/PMR-15 compression specimens, as shown in figure 8, and are currently being evaluated.

Additional repair specimens fabricated from Celion 6000/LARC-160 are being evaluated by Rockwell International under contract NAS1-16448, "Development, Demonstration, and Verification of Repair Techniques and Processes for Celion-6000/LARC-160 Graphite/Polyimide Composite Structures." The types of flaws that are being repaired are representative of manufacturing defects, in-service damage, and in-service deterioration. Test specimens include flat laminates, sandwich panels, and skin-stringer panels. Figure 9 presents some results for damaged and repaired Celion 6000/LARC-160 sandwich compression panels. A 5.1-cm-diameter hole in one of the face-sheet laminates reduces the compression strength more than 50 percent. A cure-in-place scarf repair for similar damage is found to increase the compression strength to within 20 percent of the no-damage condition. Although full-strength recovery was not achieved in this first test, the results are encouraging because failure occurred outside the repair location. (See fig. 10.)

Large Subelement Specimens

The strength of the repaired Gr/E coupon specimens has been shown to approach the undamaged strength of the specimen, and these concepts are being evaluated for the repair of Gr/PI coupon specimens. The next step in the development of repair technology was to apply these repair concepts to larger Gr/E subelement specimens. Large subelement specimens were made available for repair demonstration through the NASA-sponsored Aircraft Energy Efficiency (ACEE) program. In this program, each of the three major U.S. commercial transport manufacturers are under contract to NASA to design, fabricate, and test secondary and primary composite components. The use of advanced composites has the potential to reduce fuel consumption by 10 to 15 percent. One of these components is the L-1011 composite vertical fin that the Lockheed-California Company is building. Composite parts of the L-1011 vertical fin were used to demonstrate the repair of graphite/epoxy T300/5208 subelement specimens, and some results are shown in figure 11. Repairs were made on a L-1011 spar segment, a L-1011 stiffened cover, and a typical wing-cover subelement. In each case, as with the laminate specimens, damage was simulated as a complete cut across the specimen width representing a cleaned-out damage area rejoined by the repair. The spar-segment repair consisted of a metal doubler on each side of the web, as indicated in figure 11, as well as stepped metal doublers on the spar cap. Doublers were mechanically fastened, and film sealant was placed between metal and graphite parts. The repaired spar element was tested in tension and failed at 92 percent of the undamaged-spar-subelement tension strength. Repair of the L-1011 stiffened cover and the wing-cover subelements consisted of precuring a hat-shaped patch and bonding this patch along with a cure-in-place patch on the skin side of the specimens. The ends of the specimens were potted, and then the specimens were tested in compression. The repaired L-1011 stiffened-cover subelement failed at 92 percent of the control-specimen strength. Failure occurred when the end-potting compound separated from the specimen, thus allowing a fiber-brooming end failure. However, because such a large repair efficiency was obtained, the test was considered valid. The wing-cover subelement specimen obtained a 79-percent repair efficiency and failed through the repair. The precured hat patch for the wing-cover specimen was much stiffer than that for the stiffened fin cover and, as a result, was much harder to fit properly to the hat element for adequate bonding. Ultrasonic "C" scan inspection following bonding verified the inadequate fit of the patch. As a result, a cure-in-place repair on the hat element as well as on the skin is recommended.

Plans for Future Work

Repair techniques have been successfully demonstrated on small coupon specimens, and selected repair techniques have also been shown to be applicable to the repair of larger subelement specimens. However, it must be demonstrated that the methods developed are suitable for repair of large-area damage on full-sized structures and, as a result, the L-1011 composite vertical-fin ground-test article (GTA) has been selected for evaluation. The GTA is a full-sized production component and is a complete box structure, approximately 7.6 m by 2.7 m in area, with composite hat-stiffened covers and composite spars and ribs. (See fig. 12.) The damage to be inflicted will simulate a swept-stroke lightning strike near the bottom of the fin as shown in the figure. An area of approximately 10 cm by 30 cm will be delaminated; then the skin is punctured with a hot welding rod. The delaminated area will then be exposed to a high-temperature flame. The large-area repair planned for the vertical fin consists of a cure-in-place external Gr/E patch for repair of the skin damage and a cure-in-place Gr/E patch for repair of the damaged hats. After the GTA has been repaired, the service-life capability will be determined by performing spectrum fatigue cycling. Following the fatigue cycling of the GTA, the residual strength will be determined by static loading to failure.

To address the long-term durability of repairs, a 10-year exposure program of repaired specimens will be initiated at the Langley Research Center. The specimen to be used for this program is shown schematically in figure 13 along with the proposed exposure matrix. The specimen is a tabbed 16-ply laminate specimen that is similar to specimens evaluated in tension in an earlier phase of contract NAS1-15269. The damage to be repaired will consist of a 9.5-cm-diameter hole. Four repair methods will be evaluated. These include: (1) cure-in-place external Gr/E, (2) precured bonded Gr/E, (3) cure-in-place flush Gr/E, and (4) external bolted aluminum-plus adhesive. Repaired specimens as well as undamaged and damaged unrepaired controls will be exposed outdoors for 1, 3, 5, 7, and 10 years. The exposure facility is shown in figure 14. One specimen from each group of three will be subjected to a sustained load for the entire exposure time, a second specimen will be a fatigue specimen, and the third specimen will be an unloaded specimen. This exposure program is scheduled to begin in September 1982.

CONCLUDING REMARKS

Efficient repair techniques have been developed and demonstrated on coupon and larger subelement specimens applicable to secondary graphite/epoxy structure for commercial transport aircraft. The technology base developed for repair of graphite/epoxy structures is being applied to repair technique development for graphite/polyimide structures for higher temperature aerospace applications. Future work in the repair of graphite/epoxy will address large-area repair and evaluation on a large full-sized composite component. The long-term durability of repairs of graphite/epoxy will be evaluated in a 10-year outdoor exposure program at the Langley Research Center.

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TABLE I.- SUMMARY OF NASA COMPOSITE-STRUCTURES FLIGHT SERVICE

Aircraft component	Total component	Start of flight service	Cumulative flight time, hr	
			High-time aircraft	Total component
[a]Sikorsky CH-54B helicopter tail cone	1	Mar. 1972	1 140	1 140
Lockheed L-1011 fairing panels	18	Jan. 1973	22 300	395 560
Boeing 737 spoiler	108	July 1973	23 680	1 688 330
Lockheed C-130 center-wing box	2	Oct. 1974	5 800	11 570
McDonnell Douglas DC-10 aft pylon skin	3	Aug. 1975	18 520	54 370
McDonnell Douglas DC-10 upper aft rudder	13	Apr. 1976	21 900	179 900
Boeing 727 elevator	10	Mar. 1980	6 250	57 850
Grand total	155			2 388 720

[a]Aircraft removed from service in October 1979.

TABLE II.- SUMMARY OF COMPOSITE-COMPONENT INSPECTION AND MAINTENANCE RESULTS

Aircraft component	Inspection interval, months	Inspection method	Status
[a]Sikorsky CH-54B helicopter tail cone	2	Visual; ultrasonic	Minor disbonds; no repair required
Lockheed L-1011 fairing panels	12	Visual	Minor impact damage; fiber fraying and hole elongations
Boeing 737 spoiler	12	Visual; ultrasonic	Infrequent minor damage; repaired at Boeing
Lockheed C-130 center-wing box	6	Visual; ultrasonic	No defects after more than 6 years service
McDonnell Douglas DC-10 aft pylon skin	12	Visual	Minor surface corrosion on one skin
McDonnell Douglas DC-10 upper aft rudder	3, 12	Visual; ultrasonic	Minor rib-to-skin disbond on two rudders; minor lightning strike on one rudder
Boeing 727 elevator	13	Visual	Minor lightning strike on one elevator

[a]Aircraft removed from service in October 1979.

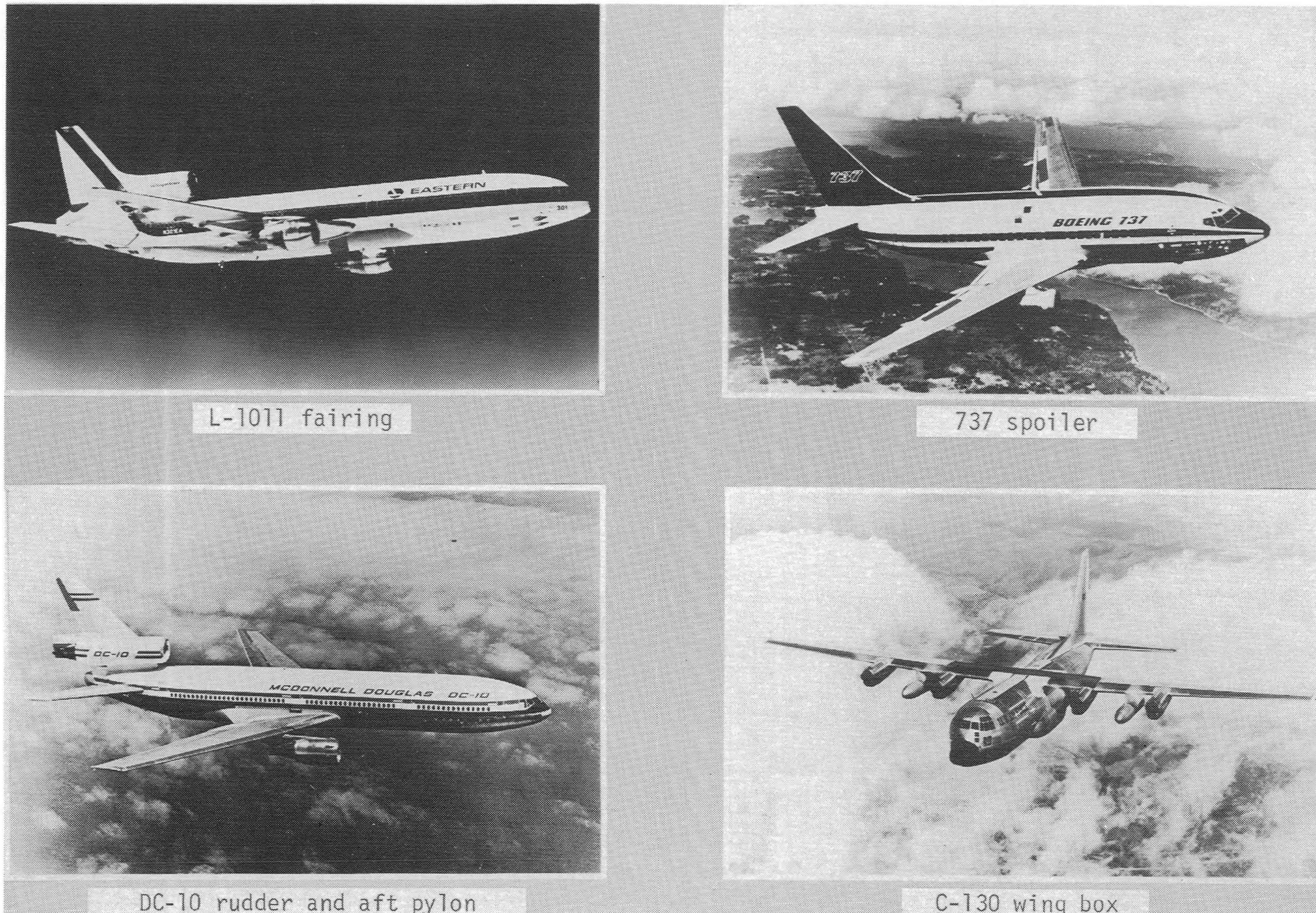

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Figure 1.- Flight-service composite components on transport aircraft.

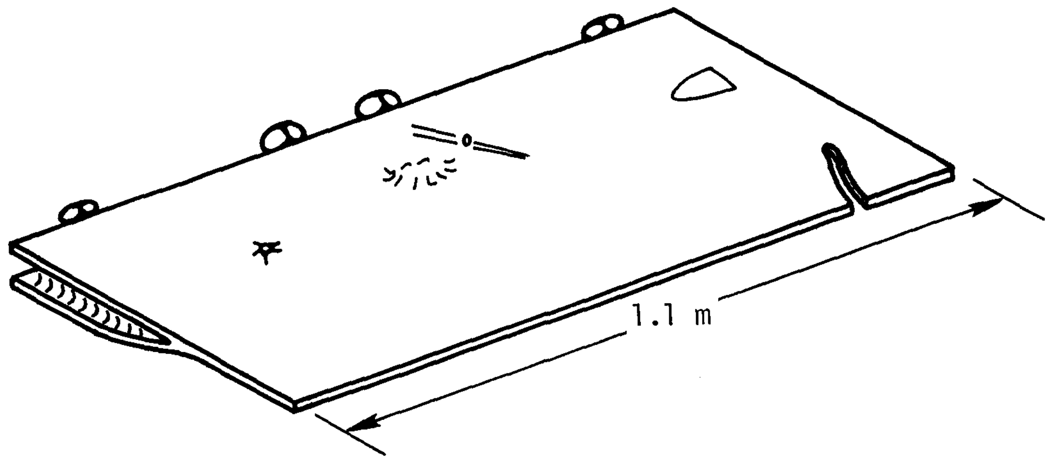
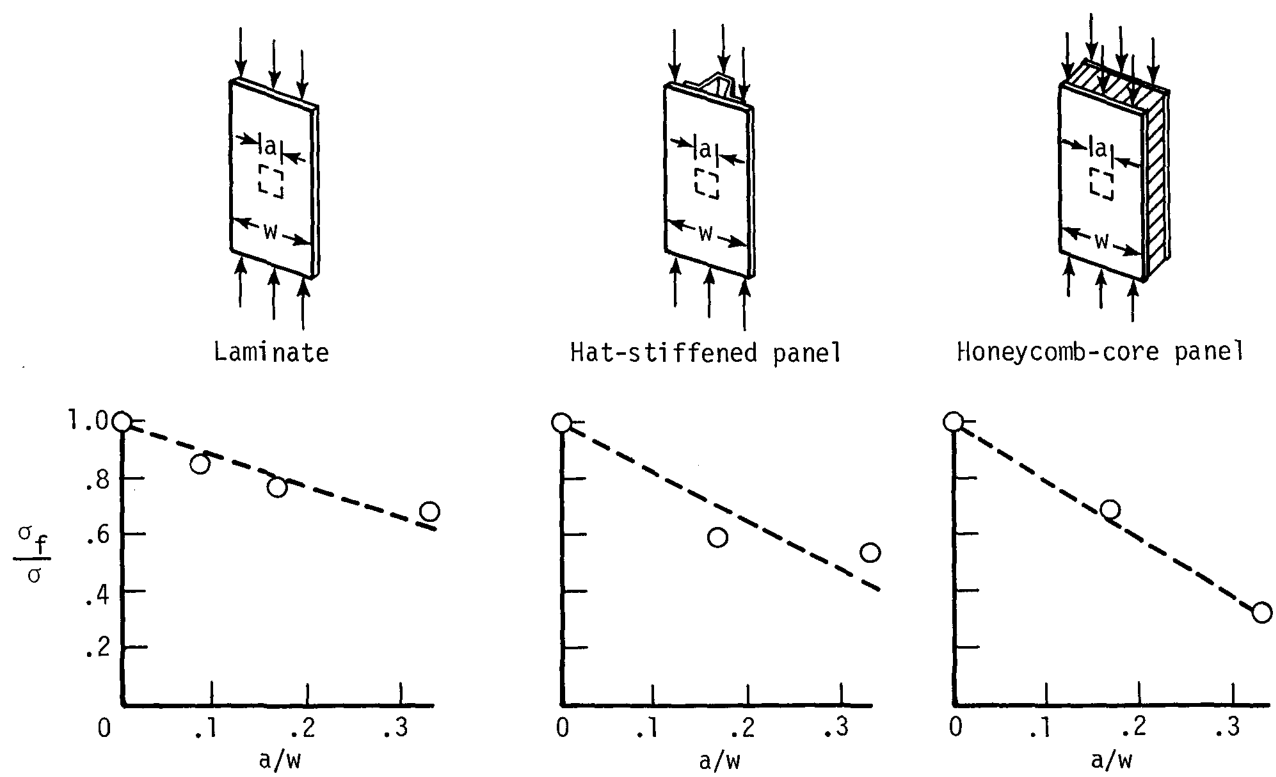


Figure 2.- Damage on graphite/epoxy 737 spoiler.



(a) T300/5208.

(b) Celion 6000/PMR-15.

Figure 3.- Effects of delamination on performance of compression panels 0.15 m wide by 0.30 m long.

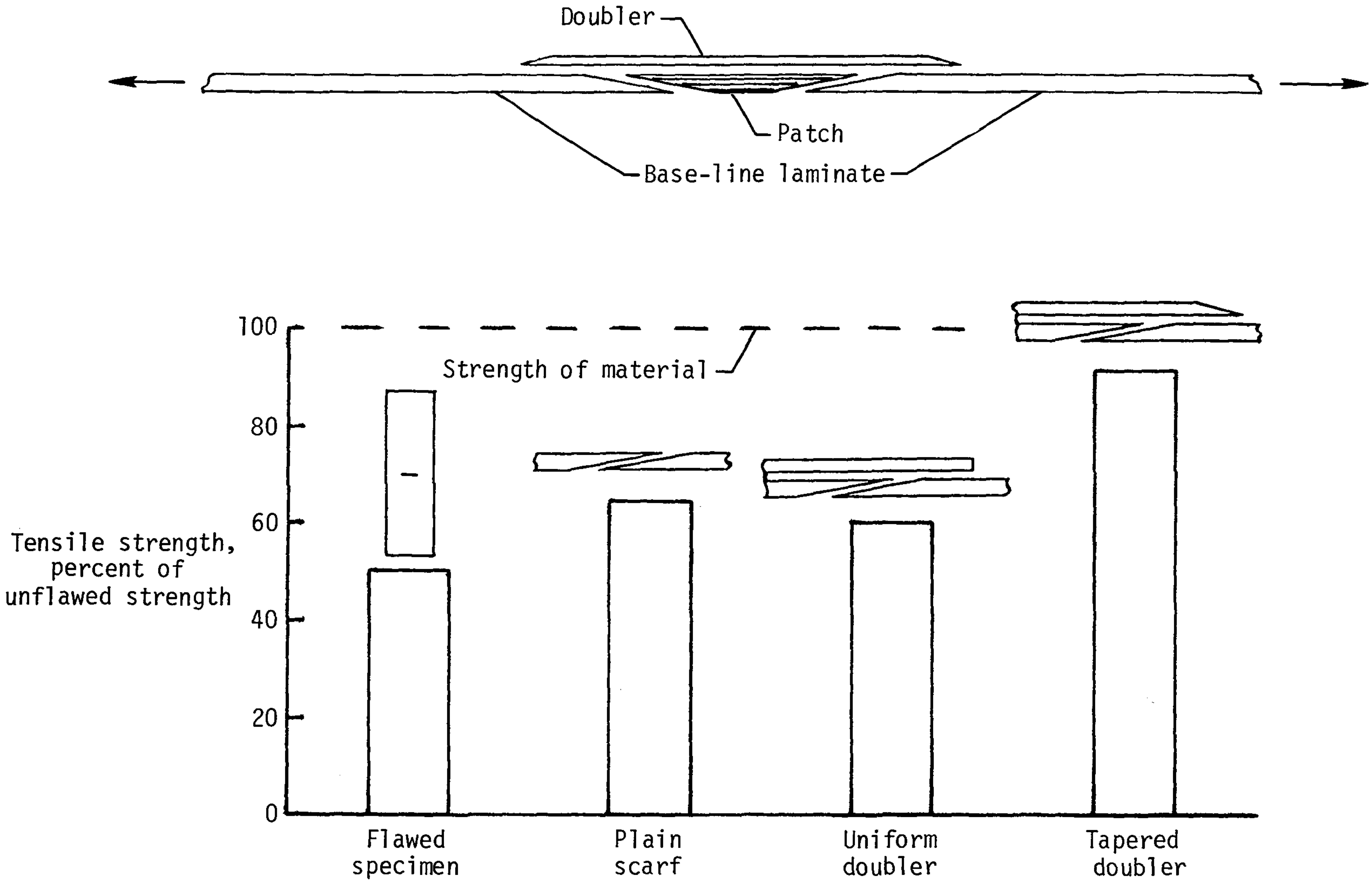


Figure 4.- Strength of repaired AS/3501-6 laminates.

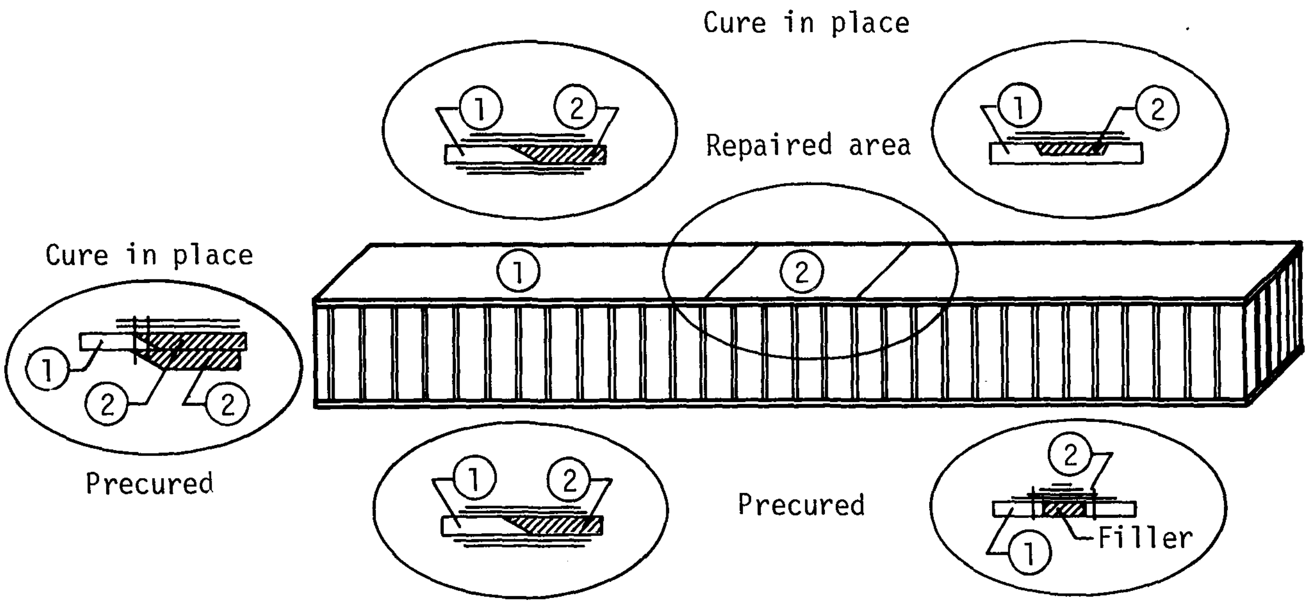


Figure 5.- Repair concepts and specimen for evaluation: ① indicates base-line laminate; ② indicates repaired area.

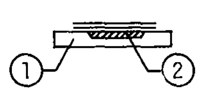
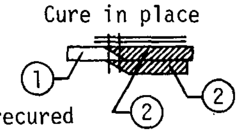
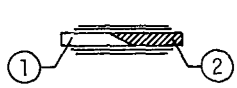
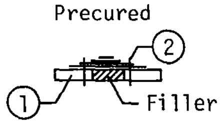
Repair concept	Repair efficiency, percent	
	Tension	Compression
Cure in place 	83	76
Cure in place 	70	65
Cure in place, Precured 	57	69
Precured 	86	90
Precured  Filler	27	19

Figure 6.- Static test results of repaired specimens. 50-ply laminate (36 percent at 0°; 56 percent at ±45°; 8 percent at 90°); T300/5208; tested at 355 K.

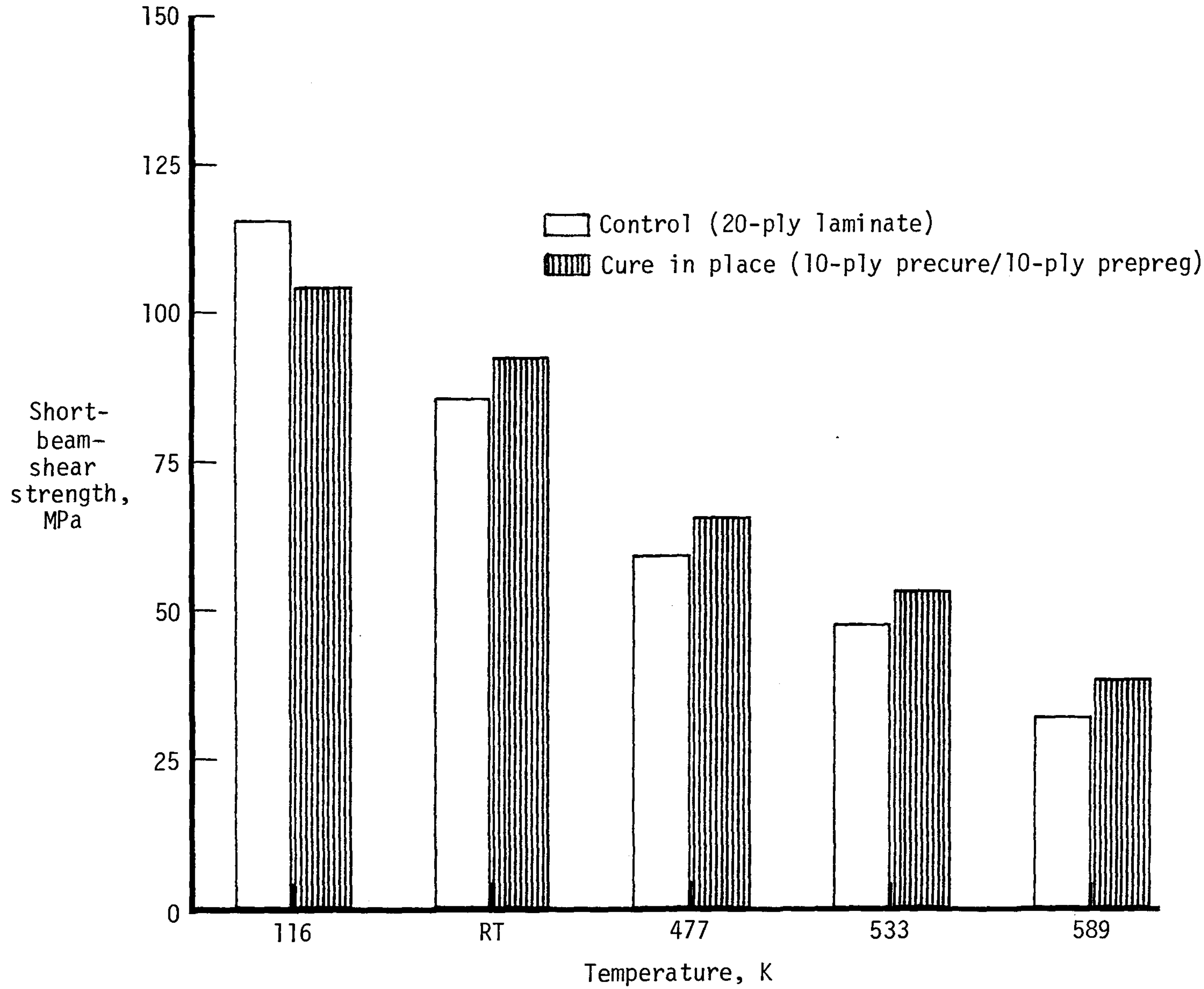


Figure 7.- Repair of graphite/polyimide Celion 6000/LARC-160.

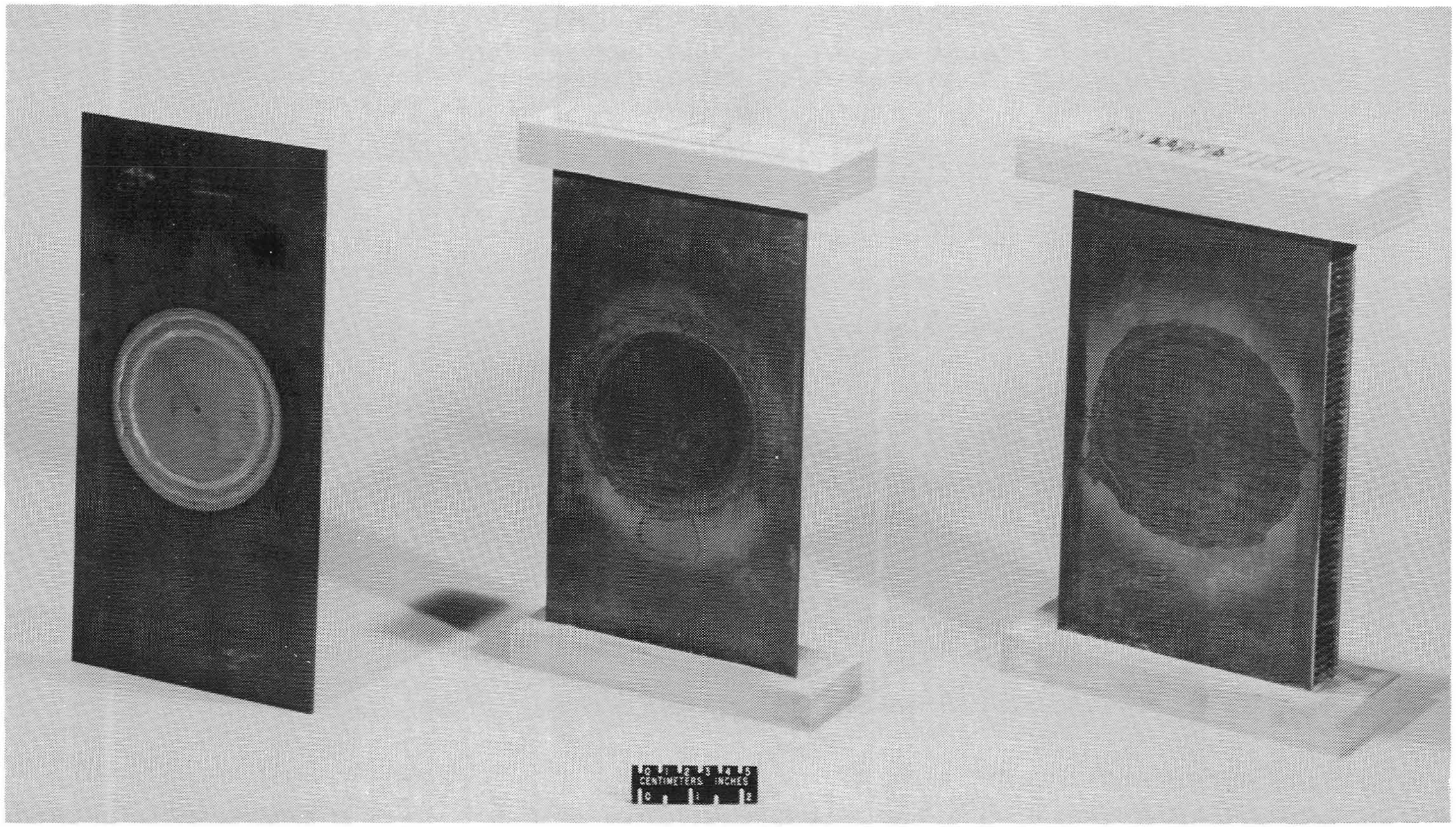


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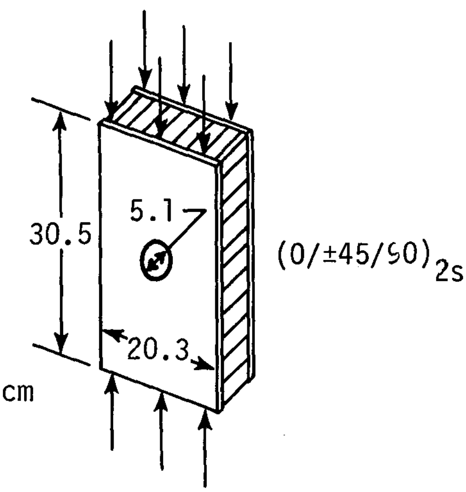
Figure 8.- Repair of Gr/PI compression specimens.



Specimen	Test results	
	Failure load, MN/m	Efficiency, percent
As fabricated, no hole	2.040	100
Damaged, 5.1-cm-diam. hole	1.007	49.4
Scarf repair of 5.1-cm-diam. hole	1.671	81.9

Figure 9.- Repair of Celion 6000/LARC-160 sandwich compression panels.

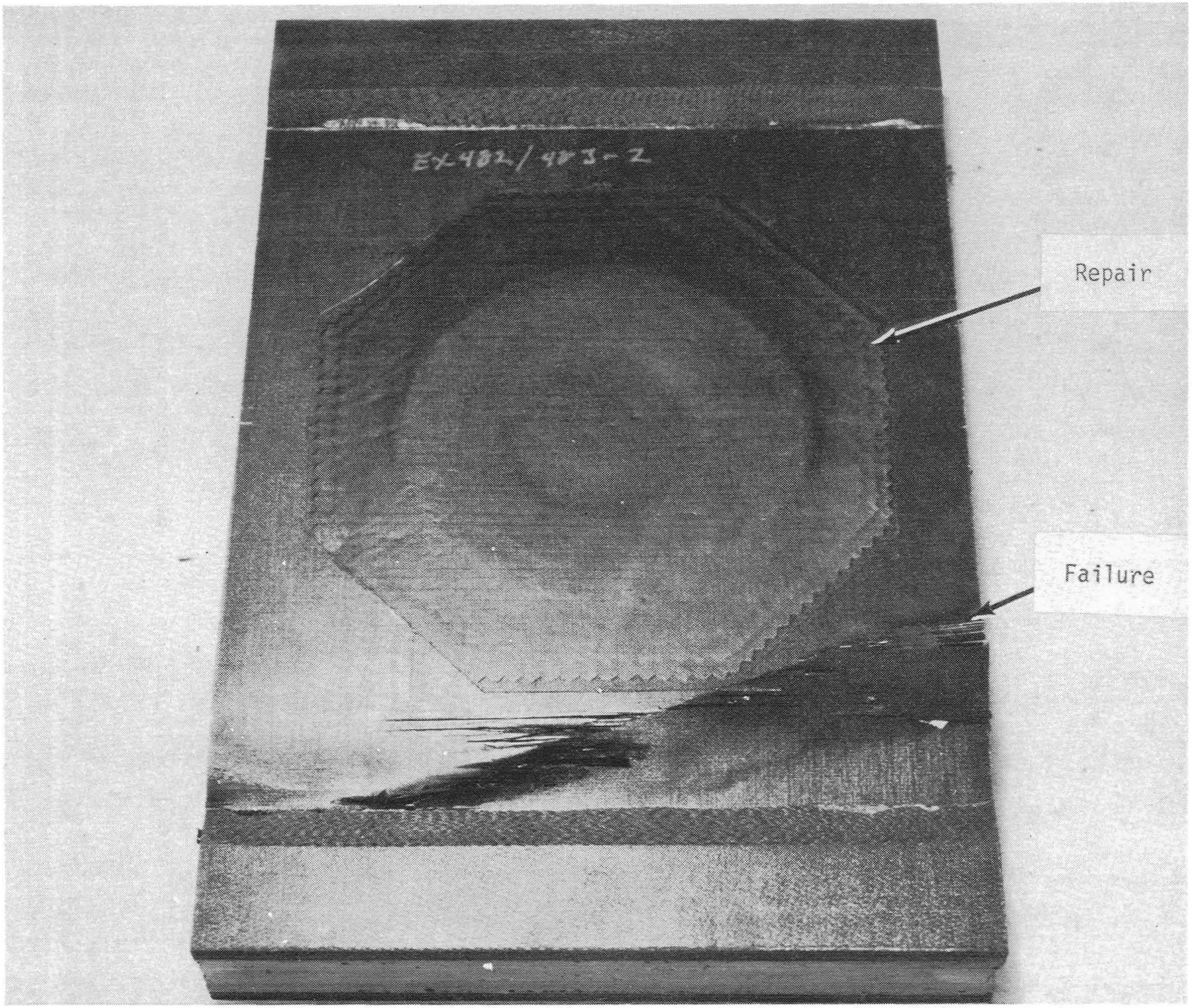


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Figure 10.- Failure location for repaired Celion 6000/LARC-160 sandwich panel tested in compression.

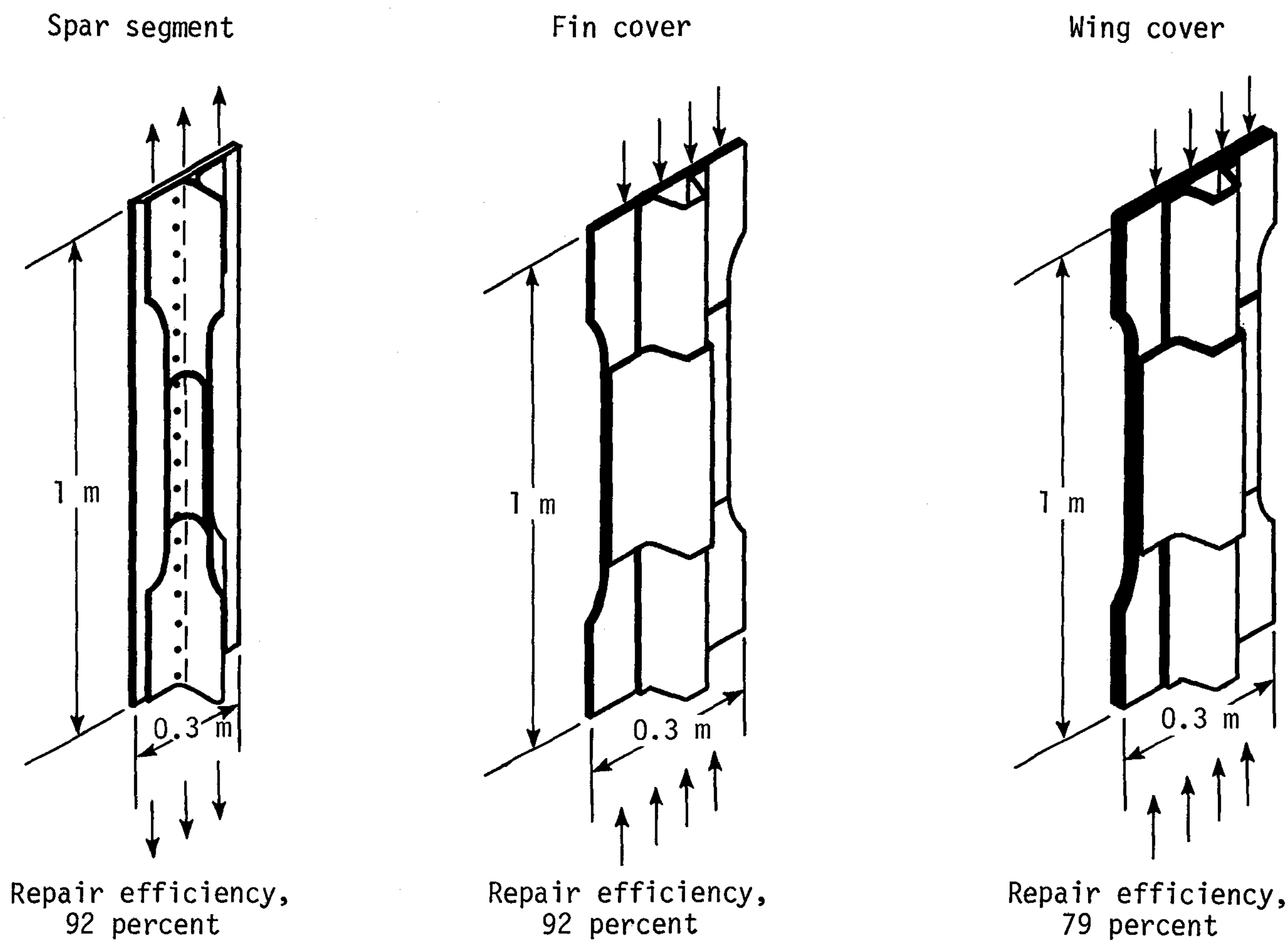


Figure 11.- Repair of Gr/E subelement specimens.

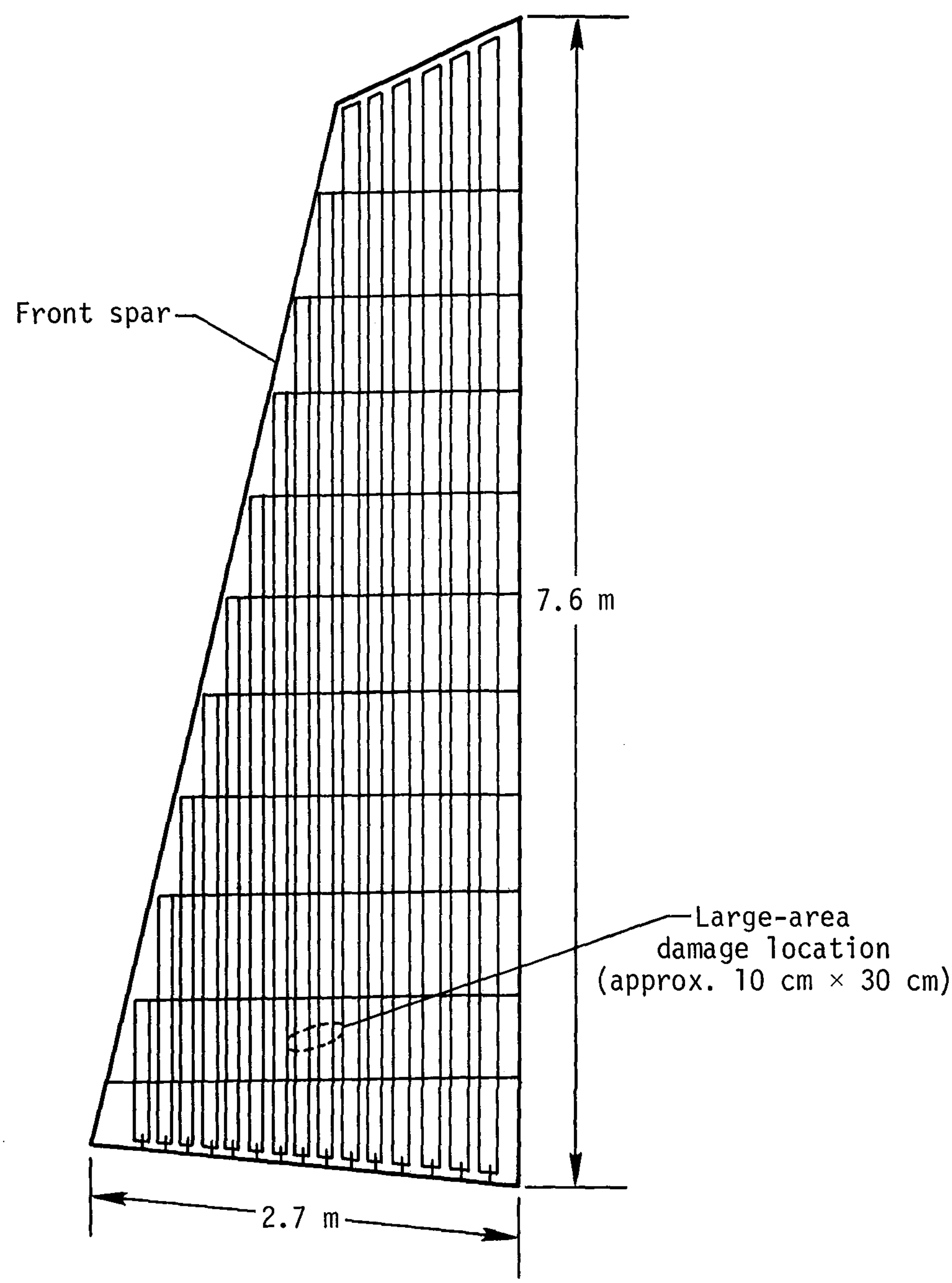
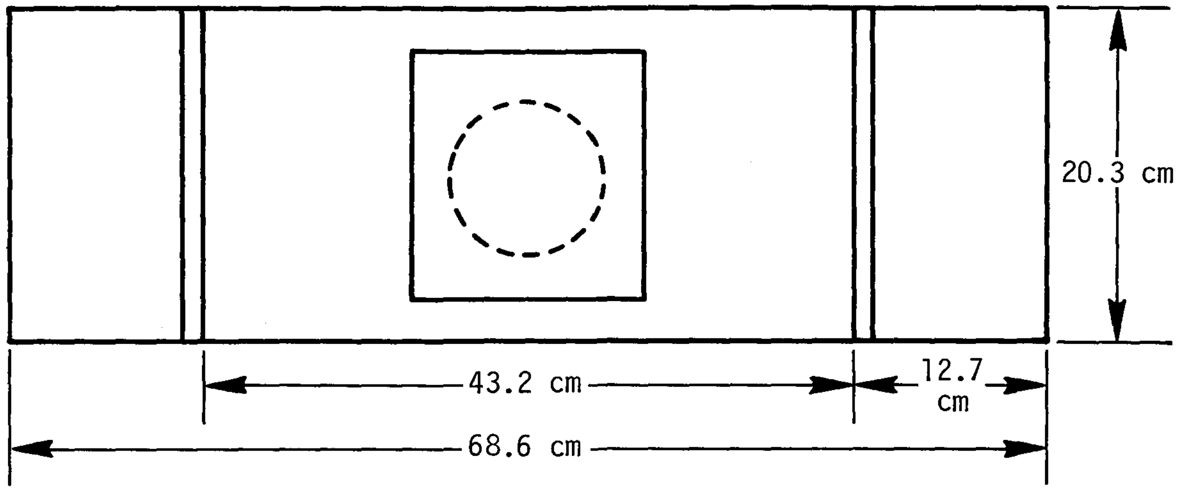


Figure 12.- Large-area damage location on composite vertical fin of L-1011 aircraft.



Exposure time, yr	Tabbed coupon specimens					
	Controls		Repair technique			
	Undamaged	Damaged, not repaired	Cure-in-place external graphite	Precured bonded graphite	Cure-in-place flush graphite	External bolted aluminum and adhesive
0	3	3	3	3	3	3
1	3	3	3	3	3	3
3	3	3	3	3	3	3
5	3	3	3	3	3	3
7	3	3	3	3	3	3
10	3	3	3	3	3	3

Figure 13.- Tabbed laminate specimens for composite-repair durability.

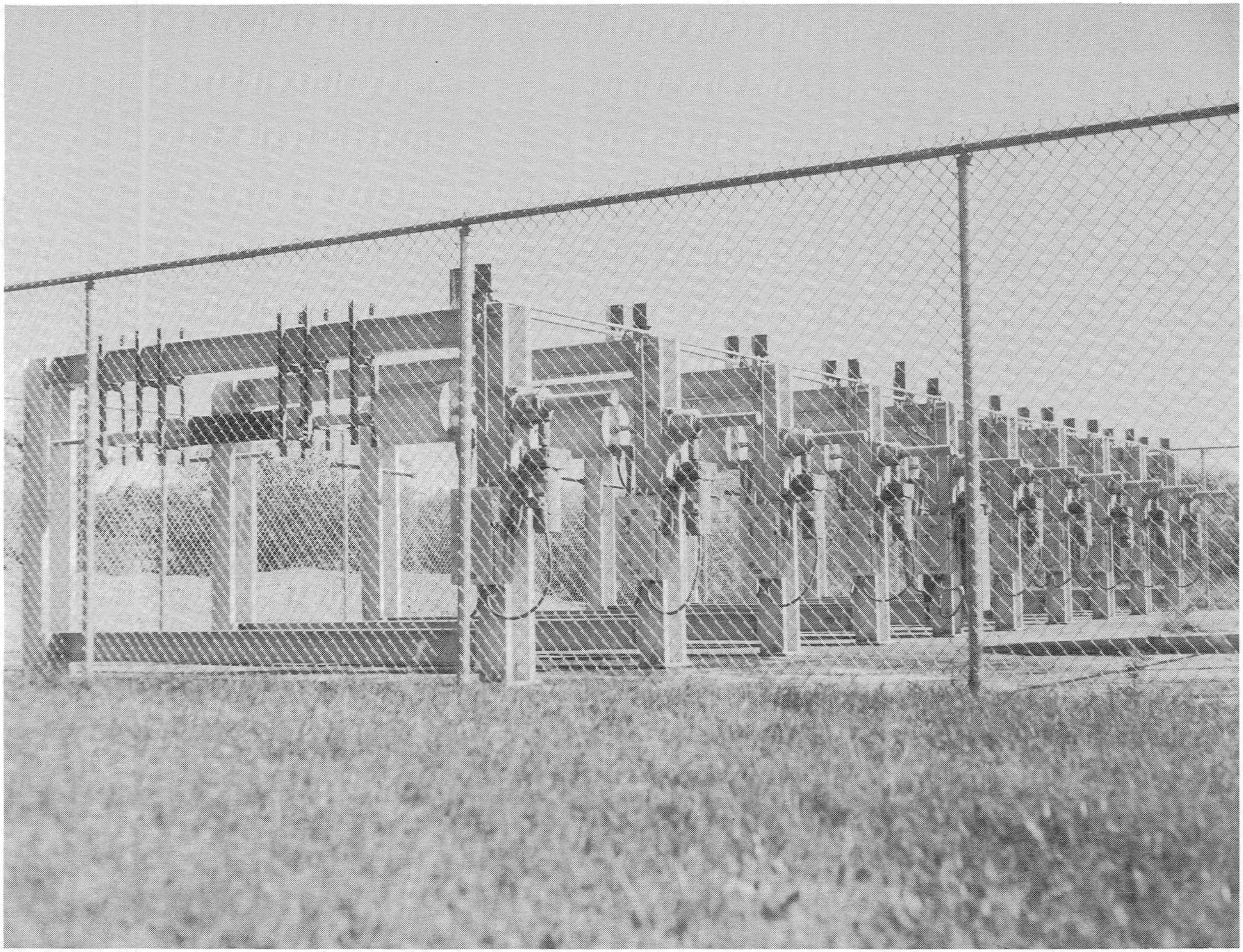


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Figure 14.- Outdoor environmental test facility.

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16. Abstract

This paper discusses the results of research for developing repair techniques for graphite/epoxy and graphite/polyimide composite structures. These techniques have been developed through Langley research, contracts with aerospace companies, and grants with universities. Tension and compression test results for several basic repair processes that were applied to damaged specimens are shown to approach the strength of undamaged specimens. Other repair configurations currently under investigation are illustrated, and future plans in the repair technology program are presented.

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