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Fuel Containment, Lightning Protection, and Damage Tolerance in Large Composite Primary Aircraft Structures

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Fuel Containment, Lightning Protection, and Damage Tolerance in Large Composite Primary Aircraft Structures

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FOREWORD

This report was prepared by the Lockheed-California Company, Lockheed Corporation, Burbank, California, under contract NAS1-16856. It is the final report of the program. The program was sponsored by the National Aeronautics and Space Administration (NASA), Langley Research Center. The Program Manager for Lockheed is Mr. A. M. James and the Project Manager for NASA Langley is Mr. H. L. Bohon. The Technical Representative for NASA, Langley is Mr. M. Dow.

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SUMMARY

This program was conducted to identify and resolve technical problems associated with fuel containment and damage tolerance of composite material wings for transport aircraft. The major tasks within the program were:

- The preliminary design of damage tolerant wing surfaces using composite materials
- The evaluation of fuel sealing and lightning protection methods for a composite material wing
- Tests of a composite wing panel designed to meet transport aircraft requirements
- An experimental investigation of the damage-tolerant characteristics of toughened resin graphite/epoxy materials.

Preliminary design studies predict that compared to the aluminum baseline, wing surfaces constructed with graphite/epoxy composites offer significant weight savings if design allowable strains can be increased from the current levels. Tests on laminates fabricated with high strain-to-failure graphite fibers combined with currently available tougher resins indicate that higher strain allowables for tension can be obtained. For greater post-impact compression strength, significant material improvements are desirable.

Based on tests conducted in this program, it is concluded that the conventional fuel tank sealing techniques used for joints in metal structures are equally applicable to composite structures. The fuel containment capability of a graphite/epoxy tank could be compromised by low energy impact damage; however, it has been determined that a 0.005-inch thick coating of a flexible polyurethane paint on the inside of the wing skin would prevent fuel leaks due to low-energy impact damage.

Swept-stroke lightning strikes to unprotected graphite/epoxy stiffened panels caused internal sparking and a large amount of structural damage. A surface protection material consisting of a graphite/aluminum wire fabric and a fastener treatment of polysulfide and a plastic cap proved effective in eliminating arcing and reducing structural damage.

The technology developed in this program was demonstrated by the fabrication and test of a blade-stiffened wing cover section including the spar-to-cover and rib-to-cover joints. The specimen test results exceeded design requirements for all test conditions.

INTRODUCTION

Current applications of composite materials to aircraft structure, most of which are stiffness critical secondary structural components and medium size primary structural components, have demonstrated weight savings from 20 percent to 30 percent. The greatest impact on aircraft performance and cost will be made when these materials are used for fabrication of primary wing and fuselage structures that are 30 to 40 percent lighter than their metal counterparts. Achievement of this goal requires innovative design concepts and improved composite materials, the performance of which must be demonstrated over a wide range of operating conditions.

In October 1981, the Lockheed-California Company began a program to identify and resolve technical problems associated with fuel containment, lightning protection and damage tolerance of composite material primary wing structure for transport aircraft. The program was sponsored by the National Aeronautics and Space Administration as part of the Aircraft Energy Efficiency (ACEE) Composites Structures Program.

This final report summarizes major technical achievements of the program. Early results from the first phase of the program are presented in References 1 and 2. Preliminary design studies conducted during the Phase I effort predicted that compared to the aluminum baseline, wing surfaces constructed with graphite/epoxy composites offer a large weight savings if design-allowable strains can be increased from the current levels of 0.004 in/in to 0.006 in/in. Tests on laminates fabricated with high strain-to-failure graphite fibers and toughened resins indicated that the desired strain allowable for tension could be obtained. Significant material improvements are required for greater post impact compression strength.

Based on the data from tests conducted during this program, it was concluded that the conventional fuel tank sealing techniques used for joints in metal structures are equally applicable to composite structures. The fuel containment capability of a graphite/epoxy tank could be compromised by low energy impact damage. Tests on impacted 0.25-inch thick graphite/epoxy laminates indicated fuel leakage even though there was no visible damage on either side of the laminate. Solutions to this problem were demonstrated.

Swept-stroke lightning strikes to unprotected graphite/epoxy stiffened panels caused internal sparking and a large amount of structural damage. Several protection systems were investigated.

The technology developed in this program was verified by the fabrication and test of a full-scale section of a blade-stiffened wing cover including the spar-to-cover and rib-to-cover joints. This structure was subjected to an extensive series of tests including; fuel pressure cycles, a swept-stroke lightning strike, one lifetime of fatigue loads, impact damage, and a residual strength test. The specimen test results exceeded design requirements for all test conditions.

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INVESTIGATION OF TOUGHENED RESIN GRAPHITE COMPOSITES

High-tensile-strain graphite fibers, in conjunction with toughened resins offer a potential to increase design-allowable strain levels. The following materials were evaluated: AS4/3502, AS4/2220-1, high-strain Celion (HSC)/982, HSC/1504 and HSC/5245C. Data on the first three materials are reported in References 1 and 2. Mechanical property tests were conducted on these materials to determine ply level properties and laminate properties. The last two materials were tested using techniques given in Reference 3 and the raw data are presented in Reference 4. A comparison of the data obtained in this program with data obtained on a wide variety of other graphite/epoxy materials is reported in Reference 5.

Ply Level Properties

Except for the HSC/982 material, tension tests were made on 0°, ±45°, and 90° laminates of the various materials and the results are shown in Table 1. The objective of these tests was to obtain the ply level properties for the materials to be used in laminate analysis. Note that the strength and failure strain of the 0° laminates fabricated using the toughened resins (5245C, 1504, and 2220-1) are considerably greater than for the untoughened resin (3502). This indicated superior translation of the fiber properties in the toughened resins. The matrix-dominated properties such as the ±45° tensile strength and 90° tensile strength are also better for the toughened resin composites.

Laminate Properties

Coupons machined from quasi-isotropic laminates were tested to investigate tension, impact damage, and compression characteristics.

Tensile tests were conducted on coupons 14.0 inches long by 2.0 inches wide as described in Reference 3. The notched specimens had a 0.25-inch diameter hole drilled through the center of the coupon. A summary of the test data obtained on AS4/2220-1, high-strain Celion/5245C and high-strain

TABLE 1. - TENSION TEST DATA COMPARISON

Laminate Orientation	Property	Material: High Strain Celion/5245C	High Strain Celion/HX1504	AS4/2220-1	AS4/3502
0^0	Resin Content	31.2%	28.4%	30.8%	28.4%
	Failure Strain (μin/in)	14200	14917	14176	11612
	Failure Stress (ksi)	295.8	314.42	299.42	236.04
	Modulus (Msi)	19.22	18.98	20.24	21.42
$\pm 45^0$	Resin Content	33.0	30.5%	29.8%	31.5%
	Tensile Failure Stress (ksi)	36.59	40.34	31.40	24.81
	Tensile Modulus (Msi)	2.65	2.85	2.41	2.65
	Shear Modulus (Msi)	0.75	0.81	0.70	0.77
90^0	Resin Content	29.6%	31.7%	30.8%	28.4%
	Failure Strain (μin/in)	7300	9342	7260	6577
	Failure Stress (ksi)	9.04	12.56	10.54	10.58
	Modulus (Msi)	1.24	1.35	1.49	1.64

Data presented is an average of 5 tests.

Celion/1504 is presented in Table 2. The data indicate that the laminate fabricated with 5245C resin has better tensile properties than the laminate fabricated with 1504 resin. Both materials have much greater failure strains for the notched coupon test than either AS4/2220-1 or T300/5208, as shown in Figure 1.

The material used for the fabrication of the L-1011 composite vertical fin and ailerons (References 6 and 7) was T300/5208. The design allowable tensile strain established for this material was approximately 0.0045 in/in for the notched (0.25-inch-diameter hole) condition. The data presented in Table 2 and Figure 1 indicate that the combination of high strain to failure of graphite fibers in a toughened resin matrix could lead to a significantly higher design allowable strain for tensile loads.

Numerous technical reports and papers published during the last five years have documented the fact that graphite/epoxy structures are vulnerable to impact damage. Test data indicate that impact damage, although not always visible, seriously degrades the compressive strength of a laminate. Impact damage could occur to a composite structure during assembly or in service and remain undetected because, in most cases, nondestructive inspections are only

TABLE 2. - QUASI-ISOTROPIC LAMINATE TENSION DATA

Material	Property	Test Condition	
		75^{o}F, Dry Unnotched	75^{o}F, Dry 0.25 in. dia. Open Hole
High Strain Celion/5245C	Resin Content (%)	31.3	31.3
	Strength (ksi)	109.1	61.1
$(45^{o}\ 0^{o}\ 135^{o}\ 90^{o})_{6S}$	Failure Strain (10^{-6} in/in)	14900	8300
	Modulus (Msi)	7.47	7.28
High Strain Celion/1504	Resin Content (%)	36.9	35.9
	Strength (ksi)	94.1	52.6
$(45^{o}\ 0^{o}\ 135^{o}\ 90^{o})_{6S}$	Failure Strain (10^{-6} in/in)	13200	7600
	Modulus (Msi)	7.17	6.97
AS4/2220-1	Resin Content (%)	32.1	34.3
	Strength (ksi)	101.9	49.3
$(45^{o}\ 0^{o}\ 135^{o}\ 90^{o})_{6S}$	Failure Strain (10^{-6} in/in)	13200	6700
	Modulus (Msi)	7.13	7.33

Data presented is an average of 3 tests.

conducted following cure of the part. Thus, design-allowable strains for graphite/epoxy must be reduced to account for the possibility of nonvisual impact damage within the structure. As a part of this program, the impact characteristics and post-impact compression properties of toughened resin composites were evaluated to assess potential improvements in design-allowable strains.

Quasi-isotropic panels, 48 plies thick were fabricated with each material and subjected to impact tests. For these tests, a 25- by 7-inch portion of the laminate was clamped to a steel plate with a 5- by 5-inch opening. The panel was struck in the center of the opening with a 12-pound impactor which had a 0.5-inch hemispherical diameter hardened steel tup. After impacting, the panels were inspected visually and ultrasonically to ascertain the amount of damage. Figure 2 presents the damage area versus the impact energy for several materials. In general, the panels constructed with the toughened

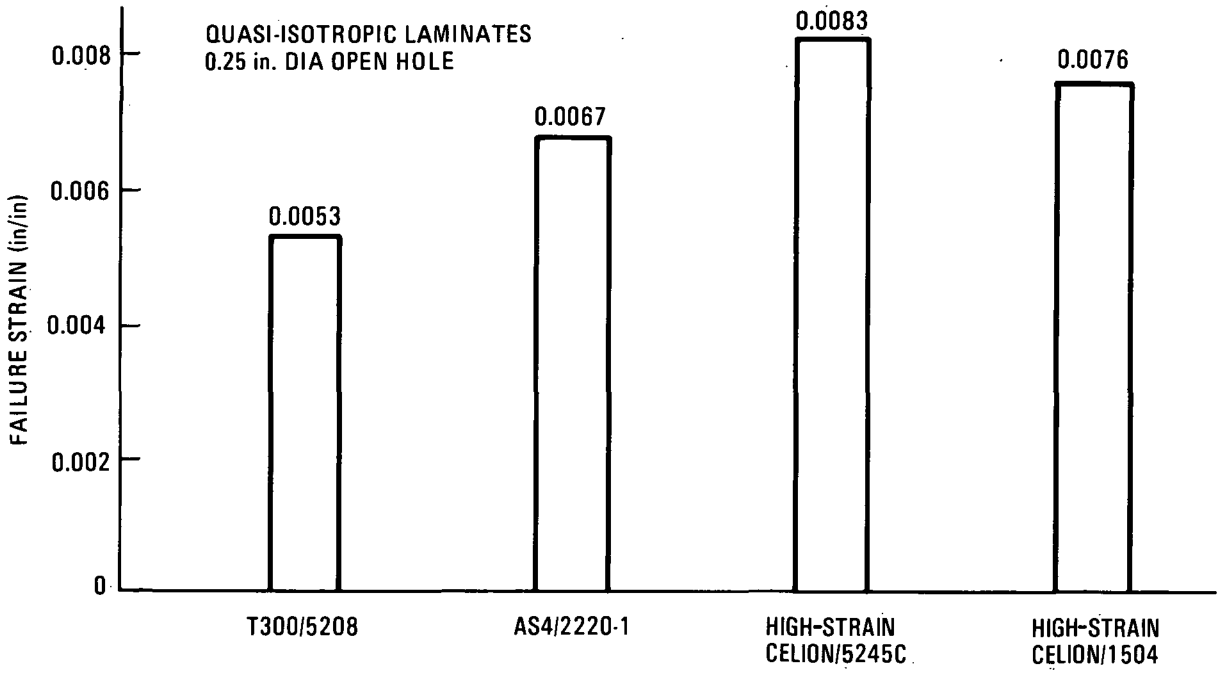


Figure 1. - Comparison of the tension failure strain of various materials.

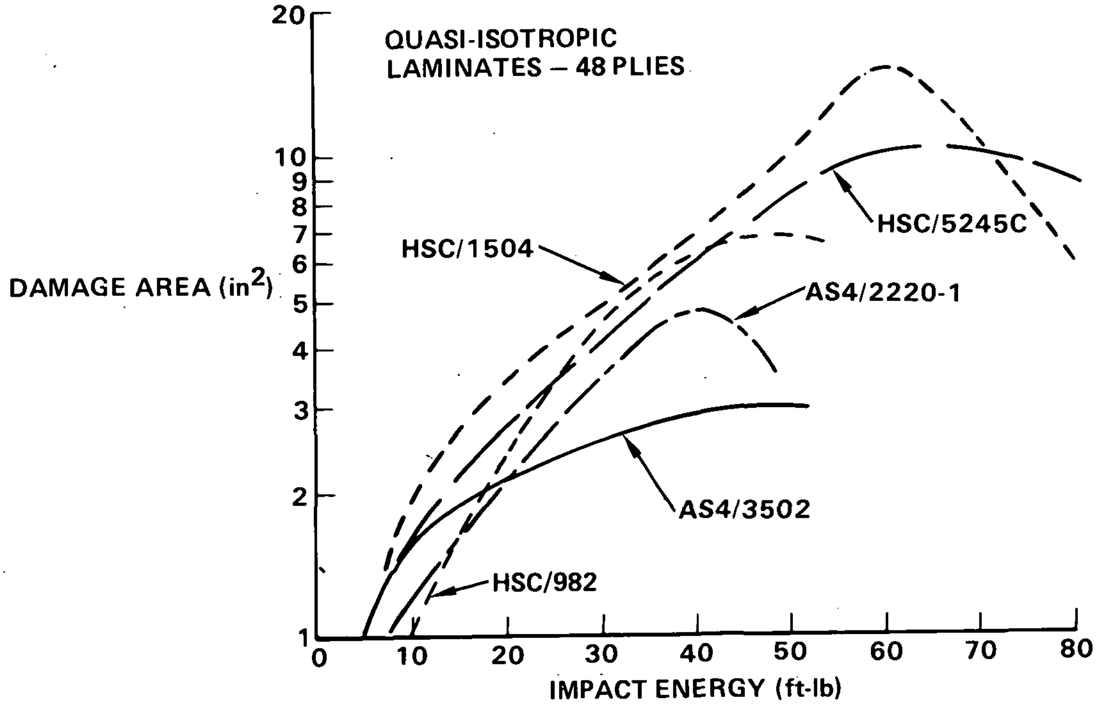


Figure 2. - Evaluation of toughened resin composites-impact response.

resins had greater damage areas than did the panel constructed with the untoughened 3502 resin. All panels displayed visual damage on the impacted side of the panel at energy levels of 20 ft-lb and greater. At energy levels of 50 to 60 ft-lb, the panels were punctured by the impactor.

Compression tests were conducted on specimens machined from quasi-isotropic laminates. Three test conditions were used for tests conducted at 75°F, (1) unnotched, (2) 1.0-inch-diameter open hole, and (3) impacted. A fourth condition, unnotched, tested at 180°F after moisture conditioning, was used to determine potential degradation of compressive properties due to environmental conditions. A summary of the test data is presented in Table 3.

A comparison of the post-impact compression strain at failure for the various materials evaluated in this program is displayed in Figure 3. For the tests conducted at an energy level of 20 ft-lb, the failure strain of the toughened systems was, in some cases, much better than the baseline 3502 material. At the 30 ft-lb energy level, the improvement in failure strain of the toughened systems over the baseline material was minimal. This can be attributed to the greater amount of damage in the toughened systems compared to the baseline material. Based on these coupon data, it would appear that none of the toughened materials offers major improvement in impacted compression strain-to-failure. Therefore, to substantially increase design allowable compression strains in structures will require improved materials and innovative design approaches.

FUEL CONTAINMENT

The use of advanced composites as the material system for primary wing box structure that contains fuel raises questions as to the integrity of the structure relative to fuel containment. The potential sources of fuel leaks include all mating surfaces, joints, and laminates with impact damage.

A comprehensive test program was undertaken to evaluate materials and methods to develop joint configurations that would provide satisfactory joint fuel sealing integrity. The results of those tests indicated that the conventional approach taken with metallic box structure is applicable to the composite structure (Reference 1). The validity of this joint sealing approach was demonstrated through the fabrication and test of simulated box beam structures.

A preliminary assessment of potential fuel leaks through laminates was also conducted early in this program. This assessment included a limited test program which indicated that an undamaged laminate would not leak fuel while low-level impact damaged laminates would. It is significant that the impact damage was not visually detectable on either the front or back surfaces. Additional tests were conducted to evaluate the effect of coatings to prevent fuel leakage after impact.

TABLE 3. - QUASI-ISOTROPIC COMPRESSION TEST DATA COMPARISON

Test Condition	Property	Material ③ AS4/2220-1 ②	Material ③ High Strain Celion/982 ②	Material ③ High Strain Celion/5245 ②	Material ③ High Strain Celion/HX 1504 ②
Unnotched at 75°F Dry	Resin Content	34.3%	36.3%	33.0%	33.7%
	Failure Strain (μin/in)	-13808	-12690	-16502	-15550
	Failure Stress (ksi)	-81.90	-75.28	-94.39	-87.63
	Modulus (Msi)	6.88	6.79	6.67	6.55
Unnotched at 180°F Wet ①	Resin Content	34.3%	36.3%	33.0%	33.7%
	Failure Strain (μin/in)	-3700 ④	-3543 ④	-13270	-13586
	Failure Stress (ksi)	-24.89 ④	-22.60 ④	-80.59	-71.77
	Modulus (Msi)	7.09 ④	6.82 ④	7.11	6.81
Notched at 75°F Dry 1.00 in. diameter Open Hole	Resin Content	34.3%	36.3%	33.0%	33.7%
	Failure Strain (μin/in)	-4713	-4960	-5415	-5342
	Failure Stress (ksi)	-31.09	-30.43	-35.71	-33.97
	Modulus (Msi)	6.64	6.33	6.59	6.49
20 ft-lb Impact at 75°F Dry	Resin Content	34.3%	36.3%	31.3%	33.7%
	Failure Strain (μin/in)	-4050	-5257	-4400	-4894
	Failure Stress (ksi)	-26.43	-31.68	-30.6	-31.83
	Modulus (Msi)	6.78	6.08	6.90	6.57
	Impact Damage Area (in^2)	1.92	2.46	2.97	3.52
	Impact Damage Width (in.)	1.45	1.72	1.98	2.15
30 ft-lb Impact at 75°F Dry	Resin Content	Data Not Available	Data Not Available	31.3%	32.6%
	Failure Strain (μin/in)			-3700	-4005
	Failure Stress (ksi)			-25.3	-26.14
	Modulus (Msi)			6.82	6.48
	Impact Damage Area (in^2)			4.45	5.03
	Impact Damage Width (in.)			2.43	2.57

① Conditioned in water at 160°F for 45 days.

② All laminates are $(45^0\ 0^0\ 135^0\ 90^0)_{6S}$

③ Data presented is an average of 3 tests.

④ 1.00 in. diameter open hole.

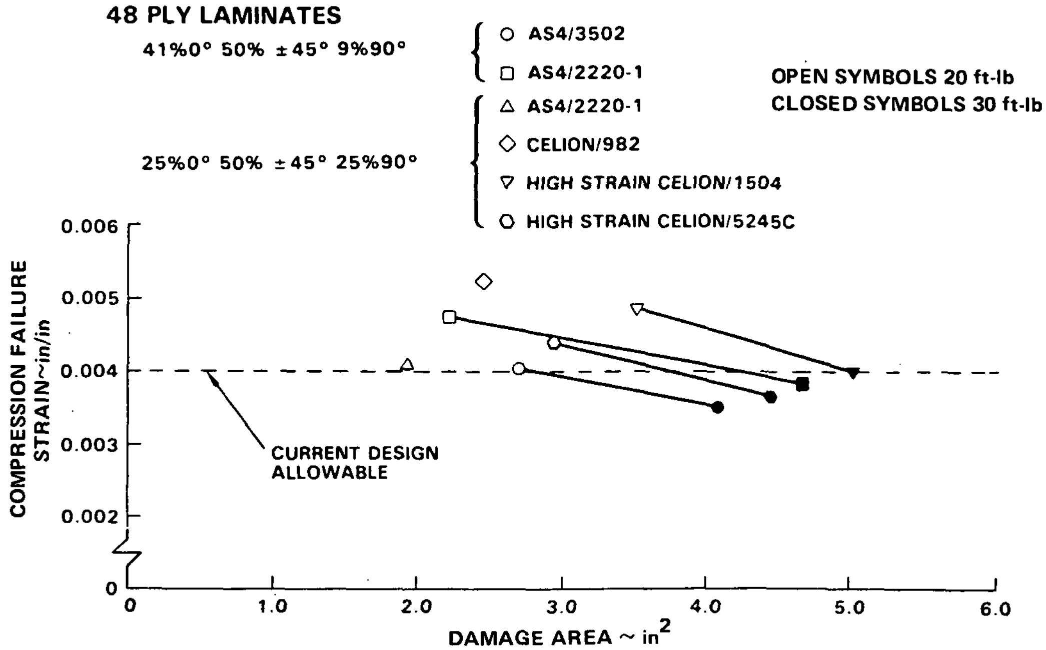


Figure 3. - Post-impact compression strain comparison.

Joint Sealing

The results of lap shear joint tests indicated that joints sealed with polysulfide met all design requirements. To verify the applicability of polysulfide sealant for complex composite structures, three box beams were designed, fabricated and tested. Two fastener types and spacings were evaluated with the three beam specimens.

The configuration of the box beams was chosen to simulate fuel leak paths and loading conditions typical of a spar cap-to-cover joint. The beam structure was enclosed to permit the application of internal fuel pressure. The test specimen was a small box beam 24 inches long, 7.4 inches wide and 2 inches deep, as shown in Figure 4. Aluminum ribs were placed at each end and at the center to provide support and for load introduction.

The first box beam specimen was used to evaluate the baseline fastener, NAS 4604U titanium screw with HL94LP stainless steel collar spaced at 1.125 inches (4.5 D). The second specimen used the same fastener and collar but with a fastener spacing of 1.50 inches (6 D). The simulated spar-to-cover joint on the third specimen was segmented into four zones to evaluate different fastener systems. A constant fastener spacing of 1.125 inches (4.5 D) was used in all zones. On the left side, the forward zone used the baseline fastener and collar, and the aft zone used a LGPL8SC-V08B titanium screw with a SLFC-MV08 stainless steel collar. The right side, forward zone, used the baseline fastener and collar installed with an NAS 1070-416 stainless steel washer under the collar, the aft zone also used a LGPL-V08B titanium screw with a

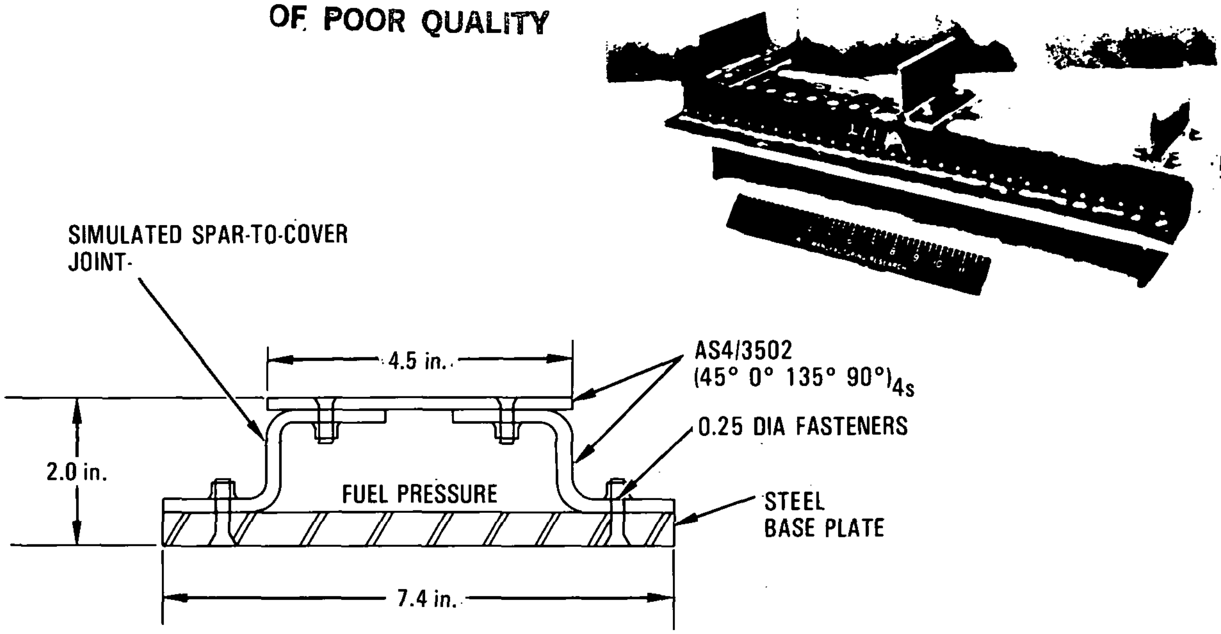


Figure 4. - Fuel sealing box beam test specimen.

SLFC-MV08 stainless steel collar installed dry, without sealant. All the other fasteners in all three specimens were installed wet (polysulfide sealant) with sealant brushed on the collars. In addition, all mating surfaces of the box had faying surface sealant (polysulfide) applied prior to assembly.

The test specimens were subjected to loads simulating the loadings experienced by a lower surface spar-to-cover joint. The box beams were loaded in three-point bending with test loads applied at the center rib location and reacted at the end ribs. Details of the test program are provided in Reference 4. The three specimens were leak-tested at 6 psi, fatigue-tested for 36,000 cycles at 50 percent limit load (R = -0.5) and 36 cycles at 30 percent limit load (R = -0.5) with 6 psi pressure maintained throughout the fatigue test. None of the specimens leaked during the leak test or fatigue tests. The three specimens were then residual-strength tested in combination with 15 psi fuel pressure. The specimens did not leak during the residual strength tests until the graphite/epoxy cover laminate ruptured in tension. A description of the three test specimens and the results of the residual strength tests are shown in Table 4.

Fuel Leakage After Impact

Tests reported in Reference 1 showed that fuel would leak through laminates after low-energy impacts that produced no visually detectable damage. Methods were investigated to prevent such fuel leaks. Two types of coating and an embedded plastic film were evaluated. The test panels, shown in

TABLE 4. - FUEL CELL BOX BEAM RESIDUAL STRENGTH TEST RESULTS

Test Specimen	Fastener Pitch (in.)	Fastener Description			*Failure Load (lb)	Fuel Leaks Prior to Failure
		Pin	Collar	Washer		
1	1.125	NAS 4604U	HL94LP	–	23,850	No Leaks
2	1.500	NAS 4604U	HL94LP	–	20,000	No Leaks
3	1.125	NAS 4604U	HL87DU	–	23,510	No Leaks
		NAS 4604U	HL87DU	NAS 1070-416		
		HUCK LGPL8SC-V08B	HUCK SLFC-MV08	–		
		HUCK LGPL8SC-V08B	HUCK SLFC-MV08	–		

*In combination with 15 psi simulated fuel pressure, Design Ultimate Load = 18,000 lb

Figure 5, were 32-ply quasi-isotropic laminates fabricated with AS4/2220-1 graphite/epoxy tape. All of the panels tested were painted on the impact surface with an epoxy primer and polyurethane topcoat.

The panels were impacted at energy levels of 10 ft-lb, 20 ft-lb, and 30 ft-lb at the locations shown in Figure 5. After nondestructive inspection, the panels were subjected to fuel-leak tests.

The leak testing of the impacted panels consisted of attaching a fuel box assembly to each impact area, pressurizing to 10 psig and then recording the time for leaking at each impact location. A fuel simulant Shell Pella-A, with a fluorescent dye additive, was used for the leak tests.

Six panels were tested. The first had no treatment and was used as a control. The second had a 0.013-inch thick polyurethane film laminated at the midplane of the 32-ply laminate. The third had a 0.021-inch thick PRC elastomeric coating on the back surface of the laminate. The fourth panel was cured with a 0.005-inch thick fiberglass fabric on the back surface and then coated with a 0.005-inch thick polyurethane based paint called Chemglaze (Lord Chemical Products). The fifth and sixth panels were also coated on the back surface with Chemglaze, 0.005-inch and 0.010-inch thick, respectively. The details of the test panels and subsequent tests are given in Reference 4.

The results of the post-impact fuel leak tests shown in Table 5, indicate that the 5-mil coating of Chemglaze is the most efficient method of controlling fuel leakage for low-energy level impacts.

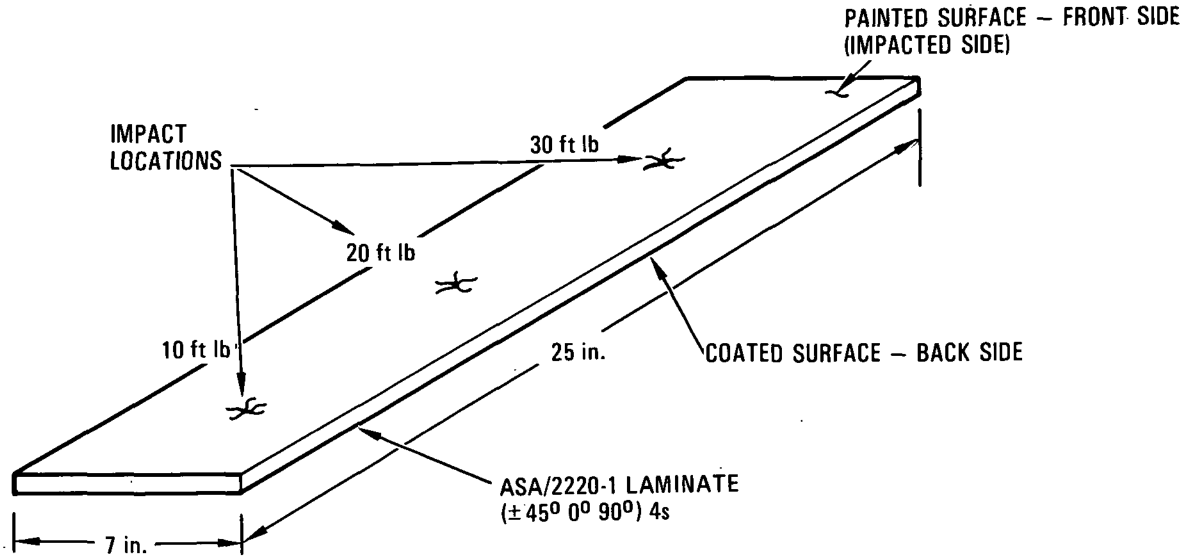


Figure 5. - Post-impact fuel leakage test laminate.

TABLE 5. - POST-IMPACT FUEL LEAK TEST RESULTS

Configuration	Coating Weight (lb/ft^2)	Time to Leak		
		10 ft-lb	20 ft-lb	30 ft-lb
Control	–	No Leak (1)	14 min., ops1	1 sec., 0 psi
Polyurethane Film at Midplane	0.080	No Leak (1)	No Leak (1)	16 min., 0 psi
PRC Elastomeric Coating	0.130	No Leak (1)	No Leak (1)	4 min., 0 psi
Fiberglass Fabric and 5 mil Chemglaze	0.098	No Leak (1)	No Leak (1)	6 min., 7 psi
5 mil Chemglaze	0.049	No Leak (1)	No Leak (1)	No Leak (1)
10 mil Chemglaze	0.098	No Leak (1)	No Leak (1)	No Leak (1)

(1) After 50 hr at 10 psi.

In addition to the above tests, another approach was evaluated, wherein a layer of film adhesive was cocured to the outer surface of the laminate. This approach has been used by other companies to achieve a smoother surface on the laminate, reduce splitting due to machining operations, and to act as a fluid barrier.

A test laminate, 32 plies of AS4/2220-1, was cocured with a single layer of FM 300 on one surface. This laminate was then subjected to impacts on the side of the laminate opposite to film adhesive. The impactor and test setup used for these impacts was that specified in NASA RP 1092, "Standard Tests for Toughened Resin Composites." Following the impact tests, the panel was visually inspected. This inspection revealed that the panel had back surface cracks at impact energies as low as 5 ft-lb. Based on the visual inspections of back surface damage, fuel pressure testing was deleted.

LIGHTNING STRIKE BEHAVIOR

A potential problem with fuel containing wing boxes constructed with graphite composites is fuel ignition due to a lightning strike. The majority of the wing box surface is classified as Zone 3 (current transfer region); however, the area behind an engine is considered a Zone 2 (swept stroke) region.

The objective of the activities in this program as related to lightning strike behavior was to determine what fastener head treatments and surface protection materials were required to minimize structural damage and eliminate sparking within the fuel cell. Stiffened panels (Figure 6) were fabricated with standard and recessed fasteners. It was anticipated that recessed fasteners would prevent internal sparking. No exterior lightning strike protection was applied to these panels.

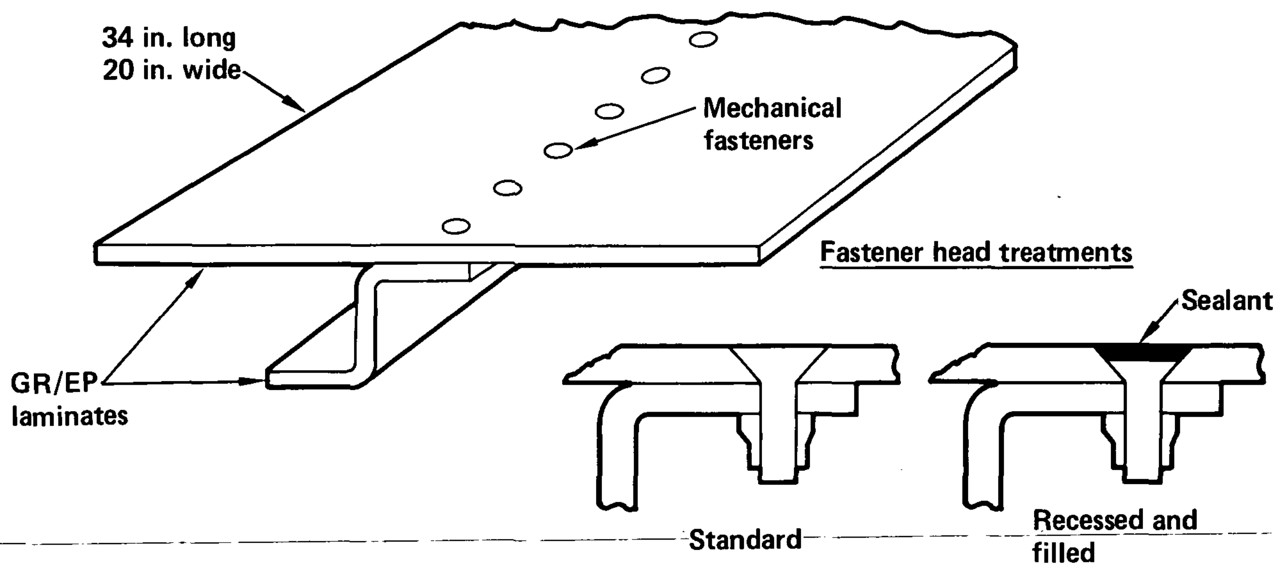


Figure 6. - Typical test panel.

The panels were tested by Lightning and Transients Research Institute for 100,000-ampere swept-stroke lightning current levels. Figure 7 illustrates the arrangement used for all of the lightning strike tests. A camera was used to determine if sparking occurred during the test. Upon the completion of the tests, the panels were inspected visually and ultrasonically to determine the amount of damage.

Results of these tests indicated that some type of surface protection would be required to prevent the substantial amount of external and internal damage which occurred due to the 100,000-ampere swept stroke. Furthermore, the recessed fastener head treatment did not eliminate internal sparking. Therefore, a variety of surface protection materials and fastener treatments were investigated.

Evaluation of Surface Protection Materials and Fastener Treatments

Four surface protection materials (Table 6) were evaluated.

The Cycom MCG material is a fabric material woven with graphite fibers that have been nickel plated. The areal weight shown in Table 6 reflects just the weight of the nickel because the graphite can be considered structural.

Two types of aluminum/graphite hybrid fabrics were investigated. The first, HMF-133AL-8/34, consists of a fabric woven with graphite yarns that

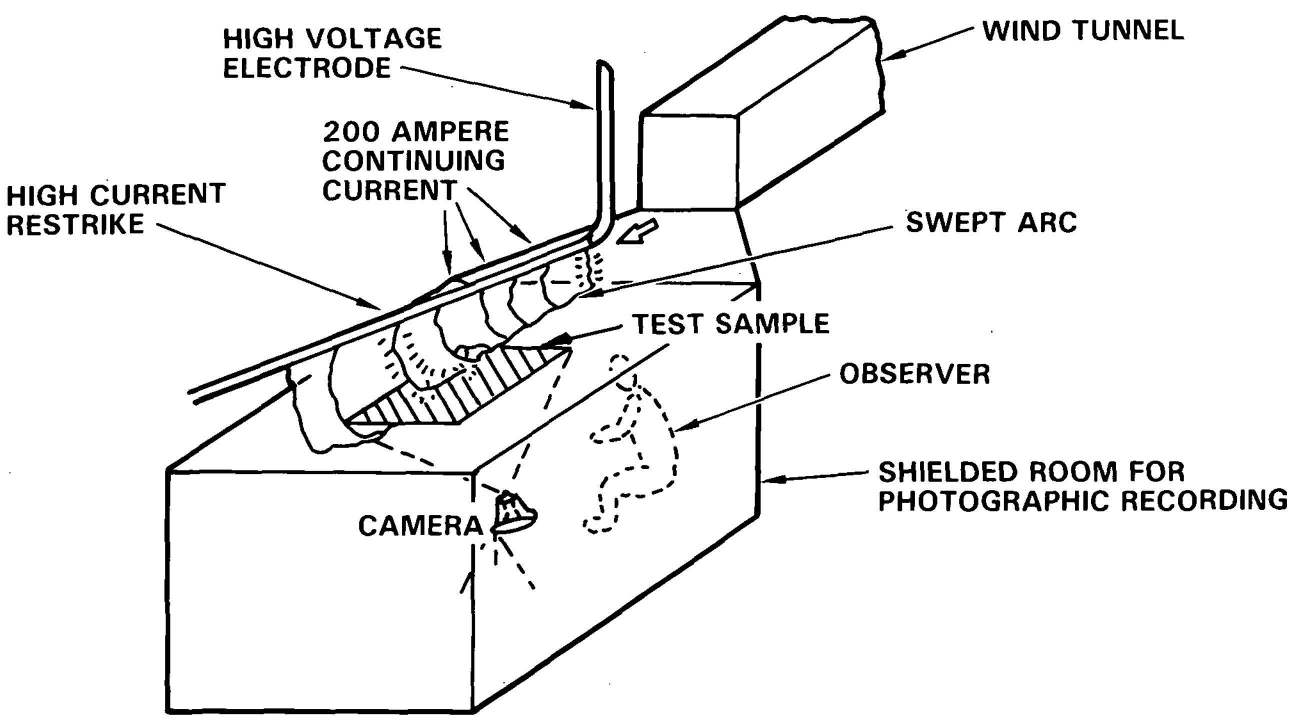


Figure 7. - Lightning strike test facility.

TABLE 6. - SURFACE PROTECTION MATERIALS

Identification	Description	Areal Weight (lb/ft^2)
CYCOM MCG	Nickel plated graphite fiber woven fabric	0.0410
HMF-133AL-8/34	4 mil dia. aluminum wire on 0.125 in. centers in woven graphite fabric	0.0029
CGS-1108	8 mil dia. aluminum wire on 0.125 in centers in woven graphite fabric	0.0117
Bostik 695-50-1	Conductive paint 0.002 in. thick, resistance = 1 ohm/ft^2	0.0427

have 4.0-mil-diameter aluminum wire wrapped around them. This results in aluminum wires on 0.125-inch centers in both the warp and fill direction. The second fabric has alternating graphite yarns and 8.0-mil-diameter aluminum wires in both the warp and fill direction. This also results in aluminum wires on 0.125-inch centers. The areal weights of these materials shown in Table 6 are for the aluminum wires only.

Another material investigated for surface protection was a conductive paint which has a resistance of 1 ohm/ft^2. This had the greatest weight of all the surface-protection material evaluated.

In addition to surface-protection materials, several fastener treatment concepts were investigated to eliminate interior sparking. The description of these concepts is shown in Figure 8.

Two panels, 262R and 264R, were tested to evaluate the effect of repairs on surface protection performance. Panels which had previously been lightning strike tested were repaired by sanding off the damaged outer ply and curing in place a patch of surface protection material, 8.0 mil diameter aluminum wire/graphite fabric (CGS-1108).

A summary of all the lightning strike test results is presented in Table 7. Note that several of the panels were struck more than once. This was accomplished by masking off adjacent areas. The information displayed for each panel includes its construction details, the number of strikes, visual observations during the test, and visual and ultrasonic post-test inspection results.

As discussed previously, the panels without surface protection sustained extensive damage due to the lightning strikes. There appears to be no correlation between panel thickness and damage. Of the various surface-protection

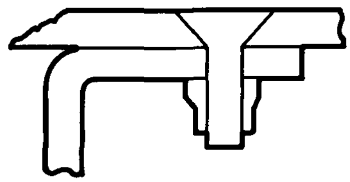
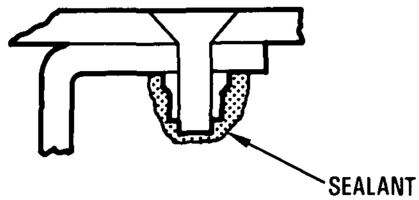
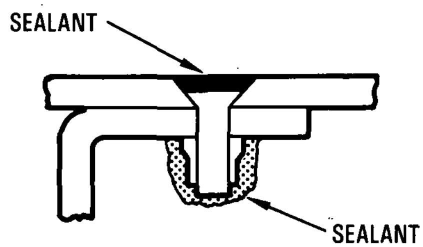
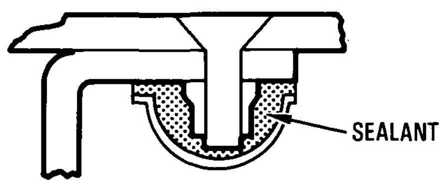
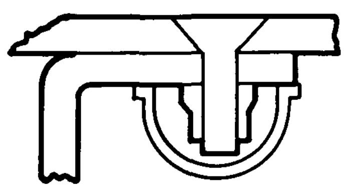
Treatment	Description
None	
Polysulfide topcoat	
Recessed and filled head and topcoat	
Topcoat and plastic cap	
Plastic cap only	

Figure 8. - Fastener treatment concepts.

TABLE 7. - SUMMARY OF SWEPT STROKE LIGHTNING TESTS TO EVALUATE SURFACE PROTECTION AND FASTENER TREATMENTS

Panel Identity	Skin Material (1)	No. Plies in Skin	Surface Protection	Fastener Treatment (2)	No. of Strikes (100,000 kA)	No. of Strikes Which Caused Visual		Average Damage Area (in^2) (3)	No. of Strikes Which Caused Sparks
						Front Surface Damage	Back Surface Damage		
27	AS4/3502	20	None	Recess and Topcoat	1	1	0	4.7	1
29	AS4/3502	28	None	Recess and Topcoat	1	1	0	13.4	1
31	AS4/3502	48	None	Recess and Topcoat	1	1	1	Not Avail.	1
26	AS4/3502	20	None	Topcoat	4	4	4	16.2	4
28	AS4/3502	28	None	Topcoat	1	1	0	5.3	1
30	AS4/3502	48	None	Topcoat	2	2	2	15.2	1
74	AS4/3502	49	CYCOM MCG	Recess and Topcoat	2	2	0	9.8	0
72A	AS4/3502	29	CYCOM MCG	Recess and Topcoat	2	2	1	8.3	1
250	AS4/2220-1	33	CGS-1108	Recess and Topcoat	3	3	0	5.4	0
253	AS4/2220-1	33	HMF-133AL-8/34	Recess and Topcoat	2	2	0	13.6	2
261	AS4/2220-1	33	CGS-1108	Topcoat	4	4	0	3.9	0
262	AS4/2220-1	33	CGS-1108	Topcoat and Cap	2	2	0	3.4	0
263	AS4/2220-1	33	CGS-1108	None	3	3	0	3.7	1
264	AS4/2220-1	33	CGS-1108	Cap Only	3	3	0	1.9	0
263N	AS4/2220-1	33	Bostik Paint	Topcoat and Cap	4	4	2	10.4	2
262R	AS4/2220-1	33	CGS-1108	Topcoat and Cap	2	2	0	Not Avail.	0
264R	AS4/2220-1	33	CGS-1108	Topcoat	3	3	0	Not Avail.	3

(1) All panels have graphite/epoxy stiffeners.

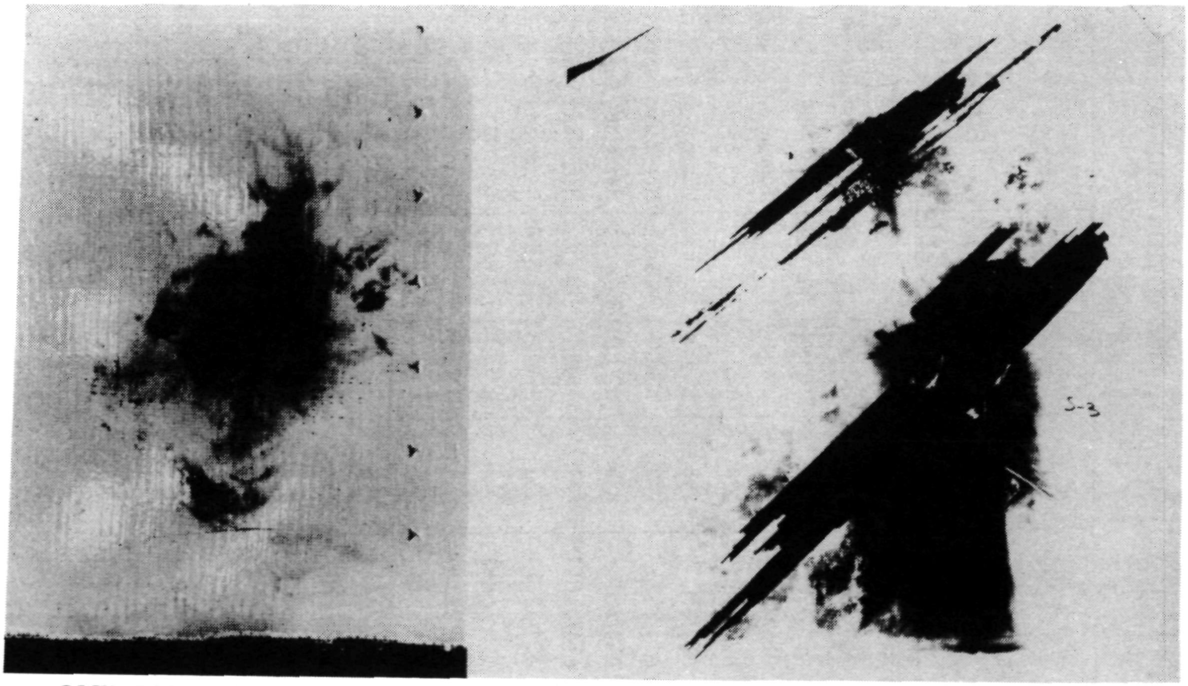
(2) 3/16 in. diameter titanium screws with CRES collars.

(3) Determined by ultrasonic inspection.

materials evaluated, the hybrid fabric containing the 8.0-mil-diameter aluminum wires (CGS-1108) performed the best. Post-strike inspections indicated that only the surface ply of the skin laminate was damaged. No internal damage was found in photomicrographic inspections. Furthermore, results from post-strike compression tests conducted on coupons machined from the damaged skin confirm that the strength of the skin laminate was not degraded. Figure 9 shows a comparison between the surface damage on an unprotected panel with a panel protected with the hybrid fabric. The photographs indicate the excellent job the aluminum wires do in charge dispersion.

Of the various fastener treatments used to eliminate sparking, the plastic cap filled with sealant proved to be most reliable. Although in some tests the panels with topcoat alone did not spark, it was observed that the slightest pinhole or thin area in the topcoat would result in a spark during the test. Note that the plastic cap by itself was sufficient to prevent sparking as demonstrated by the tests on Panel 264.

In addition to the visual and ultrasonic inspections conducted on the skin and stiffener, each panel was disassembled and the fasteners examined. This inspection reveals that the shanks of fasteners taken from areas of the panels damaged by the lightning strikes show evidence of electron discharge material removal on the shanks. Metallurgical analysis indicated metal had arced away on the shank, forming oxide-coated pits. Fasteners, several inches away from the obvious surface damage on the panel also showed evidence of



8MIL DIA. ALUMINUM WIRES

NO SURFACE PROTECTION

Figure 9. - Lightning strike surface damage.

damage. This finding indicates that in repairing damaged areas, the fasteners in the undamaged area in the vicinity of the lightning strike should be removed and inspected.

Protection of Metal Substructure

Inevitably, a wing box constructed using graphite composites will have some substructural elements fabricated with aluminum. Therefore, a series of lightning strike tests were conducted to determine if the aluminum parts must be coated to prevent sparking and what type of coatings were required.

Three graphite/epoxy panels stiffened with mechanically fastened aluminum 'Zs' were lightning-strike tested. In the first panel, the stiffener was not coated or painted. On the second panel, the stiffener had been coated with an epoxy coating. The stiffener on the third panel was painted with polyurethane paint. Results of the swept-stroke tests on these three panels indicated that painting or a coating was required on the aluminum substructure. Without the coating, the aluminum sparked in every test conducted.

TECHNOLOGY DEMONSTRATION

To demonstrate the fuel containment and lightning protection technologies developed during this program, a demonstration article representative of a stiffened wing panel of a moderately loaded area of a transport aircraft wing was designed, fabricated, and tested. The demonstration article contained all the design features developed during the program and then tested to validate the approaches taken. To verify the structural design, development tests were made.

Stiffened Panel Design

The blade-stiffened panel design (AS4/2220-1 material) for the design development tests and the technology demonstration article was based on the Lockheed L-1011 outer wing station (OWS) 188 upper surface design requirements. This location is outboard of the wing engine pylon which has integral fuel tanks and a Zone 2 lightning strike requirement. The design loads at OWS 188 consisted of an axial compression load of -12,972 lb/in, a shear load of 1804 lb/in, and an outward burst pressure of 11.46 psi. The shear stiffness requirement was 858,000 lb/in.

The aluminum design at OWS 188 consisted of discrete 2.19-inch high 'Z' stiffeners mechanically fastened to the skin, 5.23 inches apart. The graphite/epoxy design was integrally stiffened with 2.35-inch high stiffeners, 6.00 inches apart. The axial stiffness of the graphite/epoxy design was 39 percent greater than that of the aluminum design due to the lower, 4000μ in/in, design allowable compression strain of the graphite/epoxy material. Load sharing between the skin and the stiffeners of the graphite/epoxy design was 30 percent in the skin and 70 percent in the stiffeners, as compared to 70 percent in the skin and 30 percent in the stiffeners in the aluminum design. Skin shear stiffness of the graphite/epoxy design was within 2 percent of the design requirement. Including the weight of the lightning strike protection and the Chemglaze fuel tank interior coating, the graphite/epoxy design weighed 20 percent less than the aluminum design.

Details of the composite panel construction are shown in Figure 10. The stiffeners were precured and machined before assembly into the panel. At the outer surface of the panel, one 0.010-inch thick ply of 8.0-mil-diameter aluminum sire/graphite/epoxy prepreg fabric was cocured to the 21% 0°, 71% ±45°, 8% 90°, outer skin laminate. The stiffener insert was adhesively bonded to the outer skin and the inner skin. The outer and inner skins were cocured together. After the panel was cured, the top edges of each stiffener were machined to a 0.12-inch radius and two plies of 120 style fiberglass fabric were wet-laminated over each stiffener.

Design Development Tests

Prior to fabrication and test of the technology demonstration article, two blade-stiffened panels 54-inches long by 18-inches wide (Figure 10) were fabricated and then cut into smaller specimens for element, trial impact and panel tests. The specimen configurations are shown in Figure 11. Details of the test program are shown in Table 8.

Elements.- The element tests were conducted on specimens cut from one of the blade stiffened panels as shown in Figure 11. Four stiffener pull-off load specimens and four stiffener side-load specimens were tested. The stiffener fail-safe specimen was tested in rail shear. The undamaged and impact-damaged stiffener specimens were tested in compression.

The pull-off and side-load tests were conducted using a 50,000-pound MTS tensile test machine. In each case, the test load was applied along the top of the stiffener and reacted along the edges of the skin as shown in Figure 12.

The design requirement for each test was 300 pounds. Four specimens were tested in each load condition. The tests were conducted under room-temperature, dry conditions. The pull-off specimens failed at an average of 1687 pounds. The side-load specimens failed at an average of 1257 pounds. In each case failure started as interlaminar cracking in the upstanding flange of the stiffener, below the line of fasteners through the stiffener. As the load was increased, the cracks propagated down to the base of the stiffener. Final failure in the pull-off specimens occurred as a delamination of the base of the stiffener just above the bond line to the outer skin. Final failure in the side-load specimens occurred as multiple delaminations in the tapered flange of the stiffener on the side of the stiffener reacting the applied load in tension.

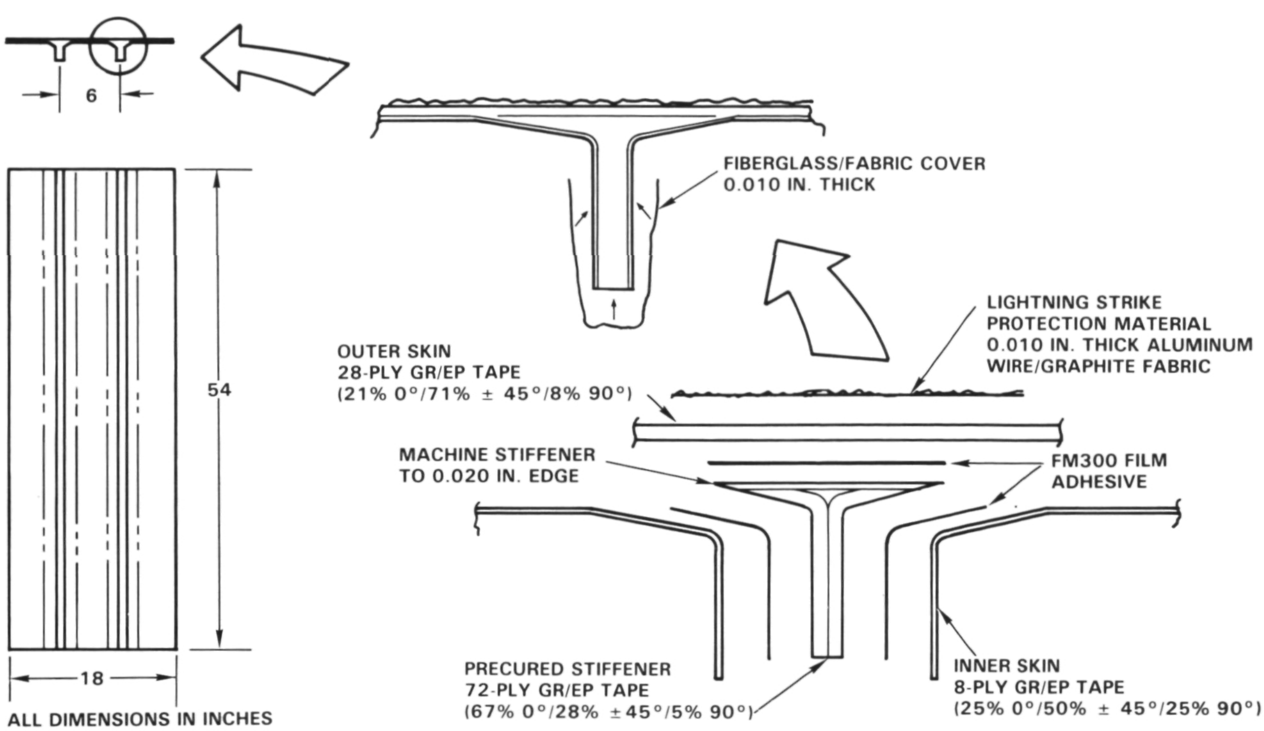


Figure 10. - Stiffened panel configuration.

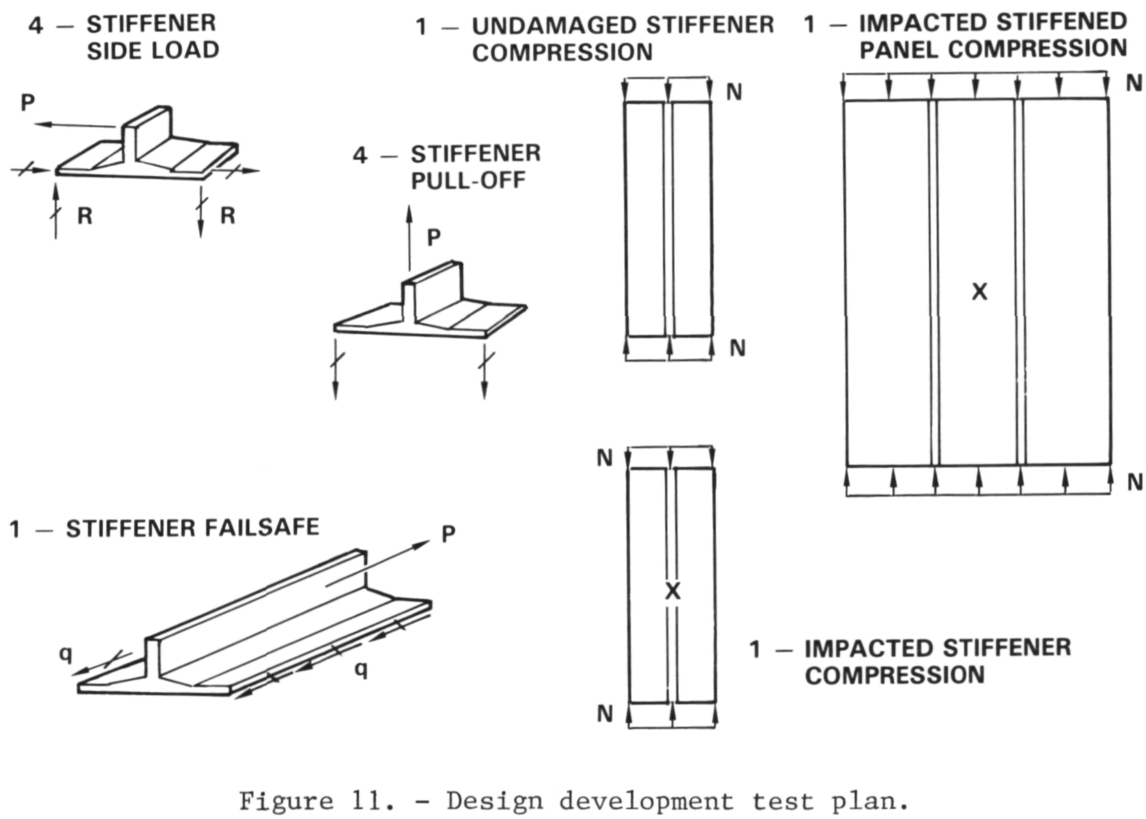


Figure 11. - Design development test plan.

TABLE 8. - DESIGN DEVELOPMENT TESTS

Specimen Description	Specimen Dimensions Length (in.)	Width (in.)	Type of Test	Number of Tests
Undamaged Stiffener	18.0	5.75	Compression	1
Impacted Stiffener	18.0	5.75	Compression	1
Stiffener Pull-Off	3.0	5.75	Tension	4
Stiffener Side Load	3.0	5.75	Tension	4
Stiffener Fail-Safe	18.0	5.75	Shear	1
Trail Impact Panel	24.75	18.0	–	1
Impacted Stiffened Panel	25.0	18.0	Compression	1

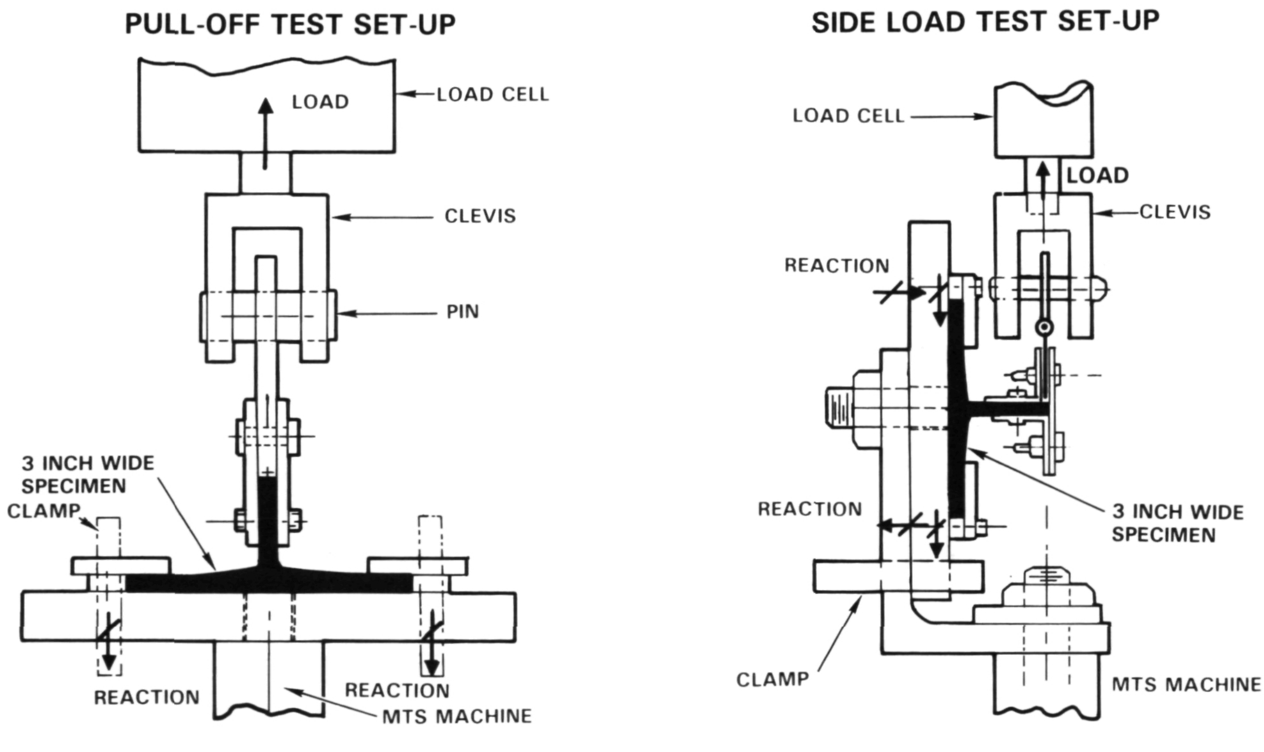
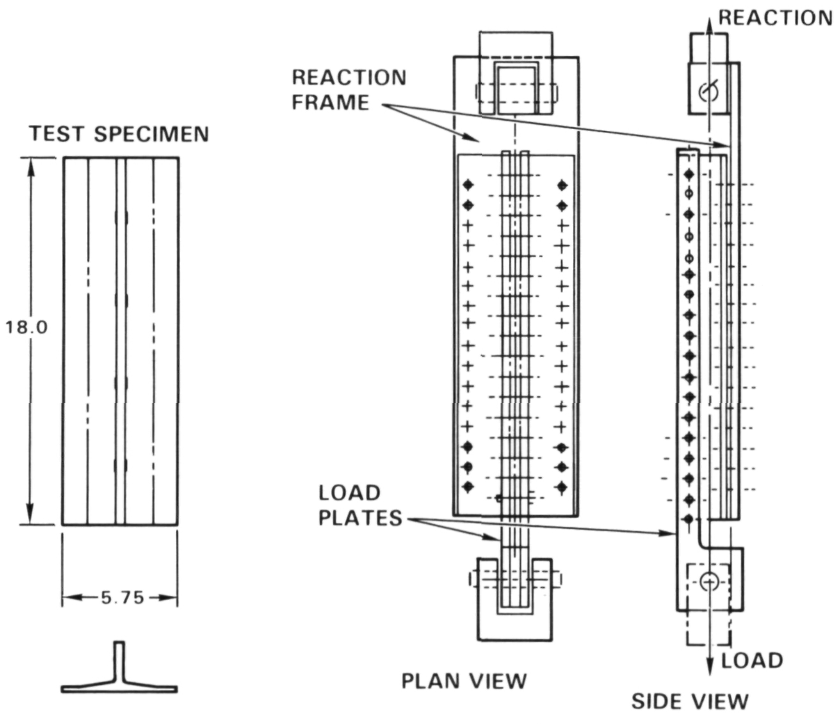


Figure 12. - Stiffener pull-off and side load test setups.

For the fail-safe test, the specimen was tested in a setup designed to load the joint between the stiffener and the surrounding panel in the same manner that it would be loaded if a stiffener in a wing were broken under load. The test was conducted under room-temperature, dry conditions. The test load was applied to the stiffener by loading plates extending the length of the stiffener (Figure 13). The load was reacted by a frame bolted to the areas of skin to either side of the stiffener.

The design requirement for the fail-safe specimen was 54,500 pounds. This was determined based on the specimen width of 5.75 inches, the design axial load intensity of -12,972 lb/in, and 70 percent of the axial load being distributed in the stiffener. The specimen failed at 57,870 pounds. The failure mode (Figure 13) was delamination of the stiffener base two or three plies above the skin/stiffener bond line.

The undamaged and impact-damaged stiffener specimens were tested in compression. The specimens were reinforced on the ends with steel boxes and potting compound. The damaged specimen was impacted with 40 ft-lb, by a 12-pound impactor having a 0.5-inch hemispherical steel tup. The impact was made in the center of the specimen on the side of the stiffener 1.25 inches from the edge of the upstanding flange. Each specimen was instrumented with back-to-back strain gages on the skin surface, on the side of the stiffener, and on the top and bottom of the stiffener (Figure 14). The tests were conducted under room-temperature, dry conditions.



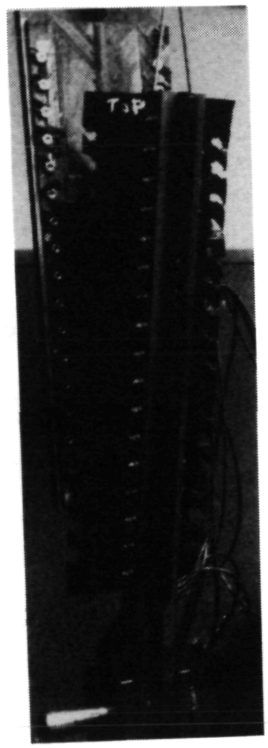


Figure 13. - Stiffener fail-safe test.

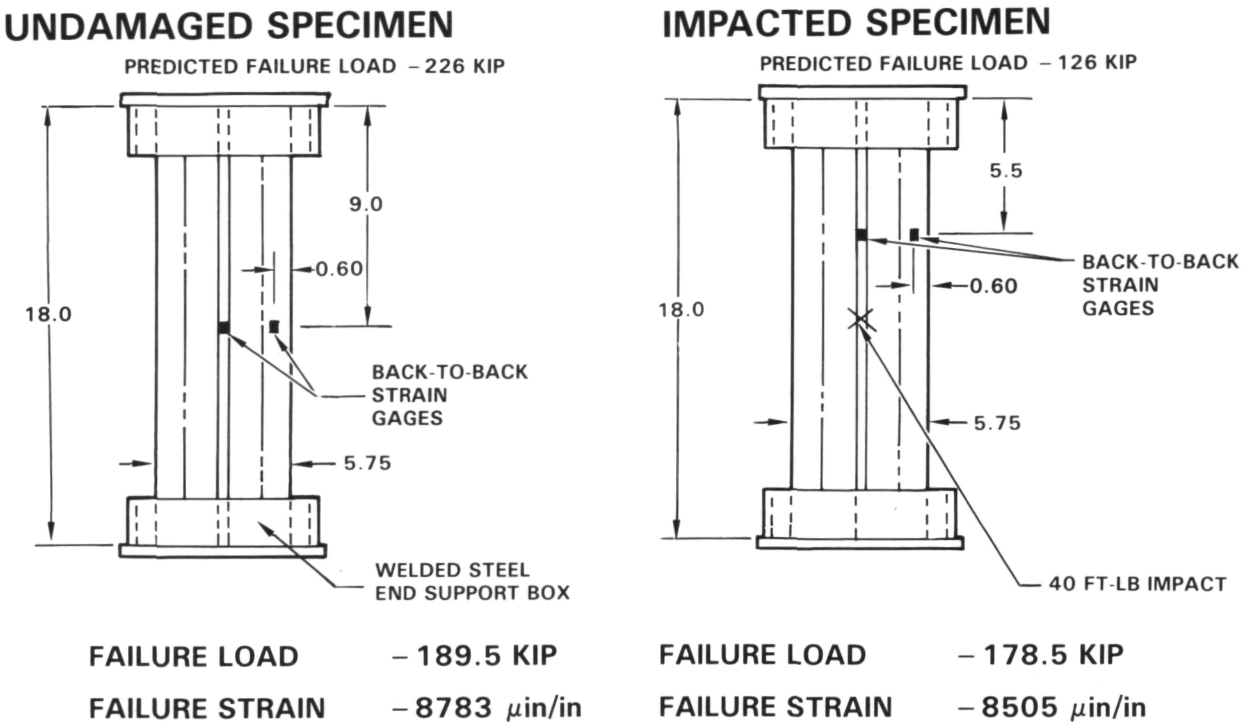


Figure 14. - Stiffener compression test set-up.

The predicted failure load was -226,000 pounds for the undamaged specimen based on the stiffener critical buckling strain of -10,500μ in/in. The specimen failed at -189,500 pounds at a maximum strain of -8,783μ in/in. The predicted failure for the impact damaged specimen was -126,000 pounds. This was based on the average failure strain of AS4/2220-1 laminates, impacted with 20 ft-lb. tested in Phase I of this program (Reference 1). The impact-damaged panel failed at -178,500 pounds at a maximum strain level of -8,505μ in/in.

Both specimens failed in combined compression and bending near the end of each specimen. The undamaged stiffener failed in the skin and in the body of the stiffener. The impacted specimen failed in the body of the stiffener near the top of the specimen as shown in Figure 15.

Trial Impacts.- One panel 24.75 inches long by 18 inches wide was used to conduct trial impact tests, to determine the level of impact energy to be used in impacting the other two-stiffener panel for the post-impact compression test. The trial impact test panel was also used to determine the impact energy that was used in impacting the single-stiffener panel discussed previously.

Twelve impacts were made on the skin surface of the panel and eight impacts on the stiffeners. Impacts on the skin areas of the panel produced barely visible front side damage at between 20 ft-lb and 30 ft-lb energy

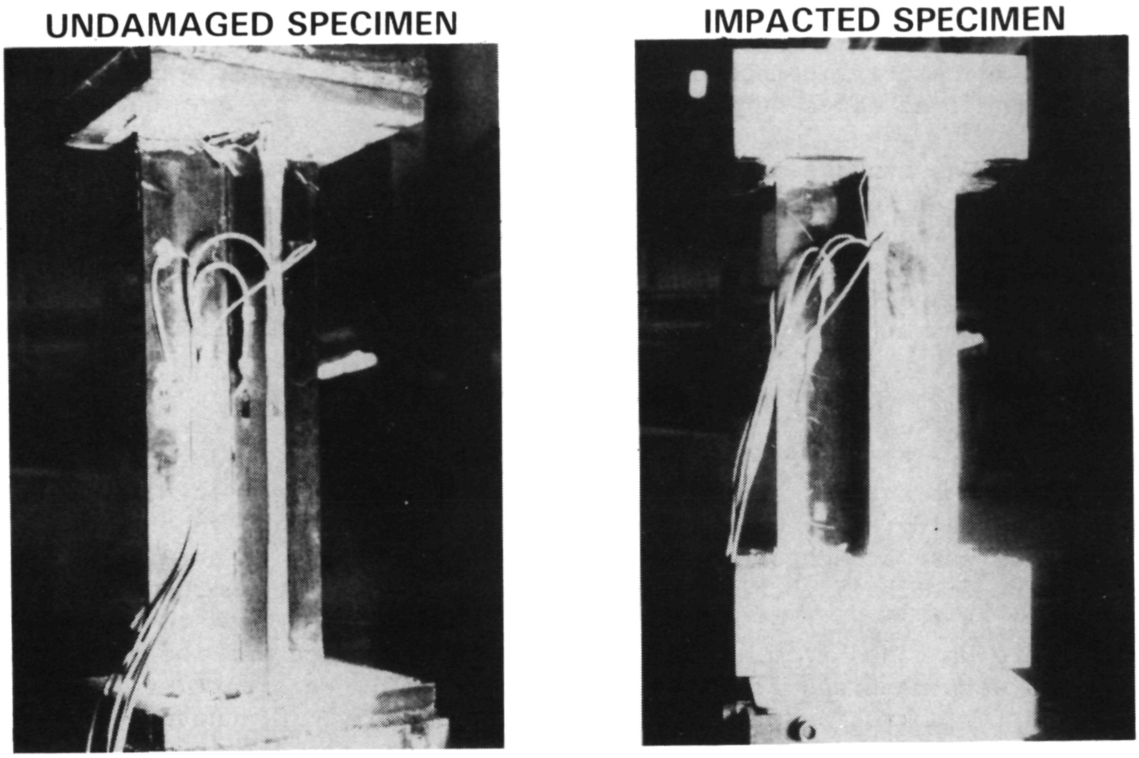


Figure 15. - Stiffener compression test failures.

levels. Visible damage to the back side of the skin occurred at 40 ft-lb energy and greater. Since the panel had a 5.0-mil Chemglaze coating on the back surface, damage that did not cause broken fibers to lift the coating was not readily visible. Stiffener impacts were made on the same panel at 10 ft-lb, 20 ft-lb, 40 ft-lb, and 50 ft-lb energy levels. Impacts on the stiffener produced barely visible damage to the fiberglass outer layer at 10 ft-lb to 20 ft-lb energy levels and visible damage at 50 ft-lb. Ultrasonic inspection of the stiffener impacts indicated that no internal damage was done to the stiffener by any of the impacts. Therefore, additional impacts were made on the panel at 40 ft-lb, 60 ft-lb, 80 ft-lb and 100 ft-lb energy levels. Visual inspection of the impacted stiffener revealed delamination of the stiffener by impacts of 60 ft-lb energy and greater. The delamination caused by the 100 ft-lb impact propagated through the other impacts on the same stiffener and delaminated 80 percent of the stiffener, as measured by ultrasonic C-scan. As a result of the trial impact test, a skin impact energy level of 30 ft-lb was chosen for the impacted stiffened panel compression test specimen, and a stiffener impact of 40 ft-lb was chosen for the impact damaged stiffener specimen.

The two-stiffener compression test panel was impacted in the center of the panel, between the stiffeners, with 30 ft-lb by a 12-pound impactor having a 0.5-inch-diameter hemispherical steel tup. The impact caused 4.3 square inches of internal damage as measured by ultrasonic C-scan. The panel was instrumented with back-to-back strain gages and the ends were reinforced with steel boxes and potting compound, as shown in Figure 16. Steel angles were clamped to the free edges during the compression test to prevent buckling.

The test was run under room-temperature, as-manufactured conditions. The panel failed in combined axial compression and bending, as shown in Figure 16, at a load of 232,900 pounds and a maximum strain of -5343μ in/in. The predicted failure load was 239,000 pounds. The failures in the stiffeners are similar to those seen in the stiffener compression tests.

Technology Demonstration Article

The demonstration article was designed to represent a moderately loaded area of a 1990s transport aircraft wing. This structure was envisioned to be made up of graphite/epoxy wing spars and covers and aluminum substructure. Aluminum ribs were attached to the panel to represent the substructure-to-surface joint for the lightning strike test and to provide chord-wise support to the panel during the fuel pressure tests and the axial load tests. The rib caps were machined from a standard aluminum extrusion and the three clips on each rib were machined from aluminum plate stock. Each part of the rib was painted for corrosion protection prior to assembly. Graphite/epoxy spars were attached to the edges of the panel to represent the spar-to-cover joint for the fuel pressure tests and to support the edges of the panel during the axial load tests. See Figure 17.

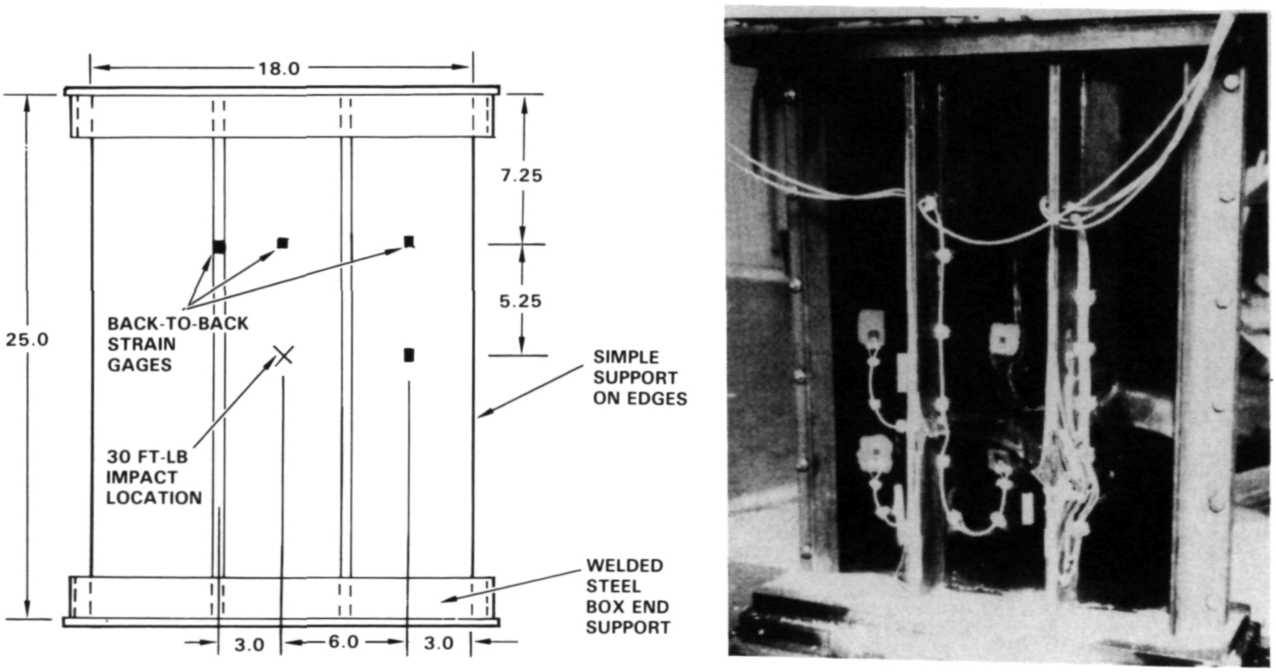


Figure 16. - Stiffened panel compression test.

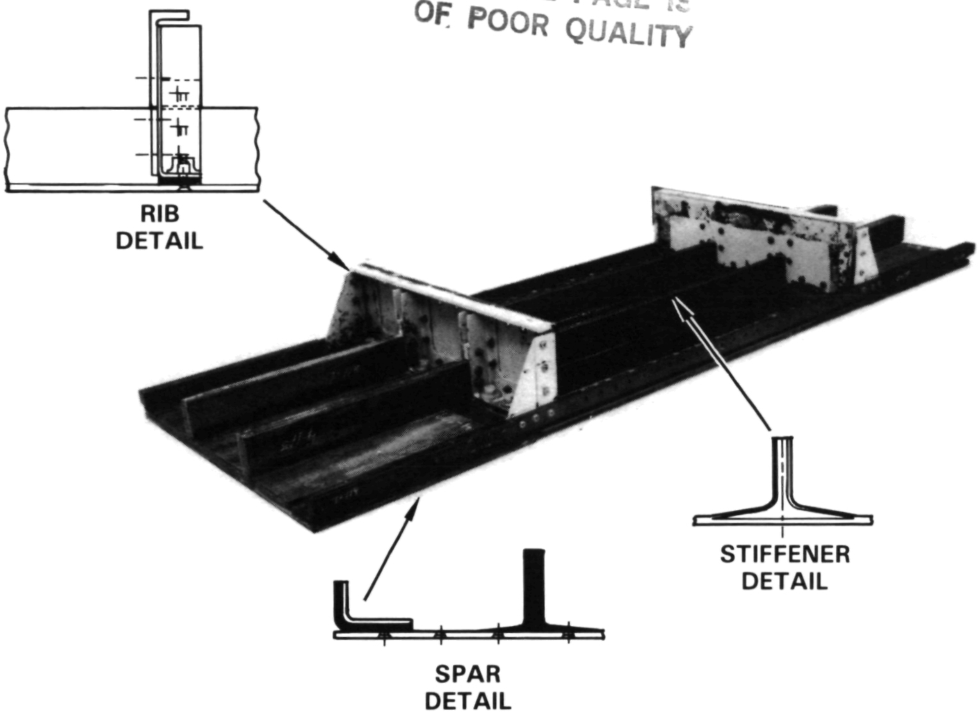


Figure 17. - Technology demonstration article.

The demonstration article was 54 inches long and 18 inches wide with two tee-sectioned (blade) stiffeners spaced 6 inches on center. The interior surface of the panel (blade side) was sealed for fuel containment using a 5.0-mil-thick flexible polyurethane coating (Chemglaze). Lightning strike protection was provided with a cocured 8.0-mil-diameter aluminum wire, graphite yarn, hybrid fabric material discussed previously. To prevent sparking, the collar of each fastener that penetrated the fuel cell was top-coated with polysulfide sealant and covered with a plastic cap. All aluminum parts within the fuel cell were painted with a polyurethane paint. The exterior (skin) surface was painted using standard epoxy primer and two coats of urethane paint (white).

Test Plan.- The testing of the technology demonstration article provided, on one component, final verification of the technology developed for lightning-strike protection, fuel containment, and damage tolerance. The test sequence is shown in Figure 18. The stiffened panel was first struck with a simulated Zone 2 lightning strike of 100,000 amperes to verify the validity of the lightning protection system. To verify the techniques developed for tank sealing, the panel was pressure-checked after the lightning-strike test and after being impacted to inflict barely visible damage to the outer surface. The damage tolerance of the panel and attached substructure was evaluated by applying one lifetime of axial fatigue ground-air-ground load cycles to the panel and then loading the demonstration article to failure in compression.

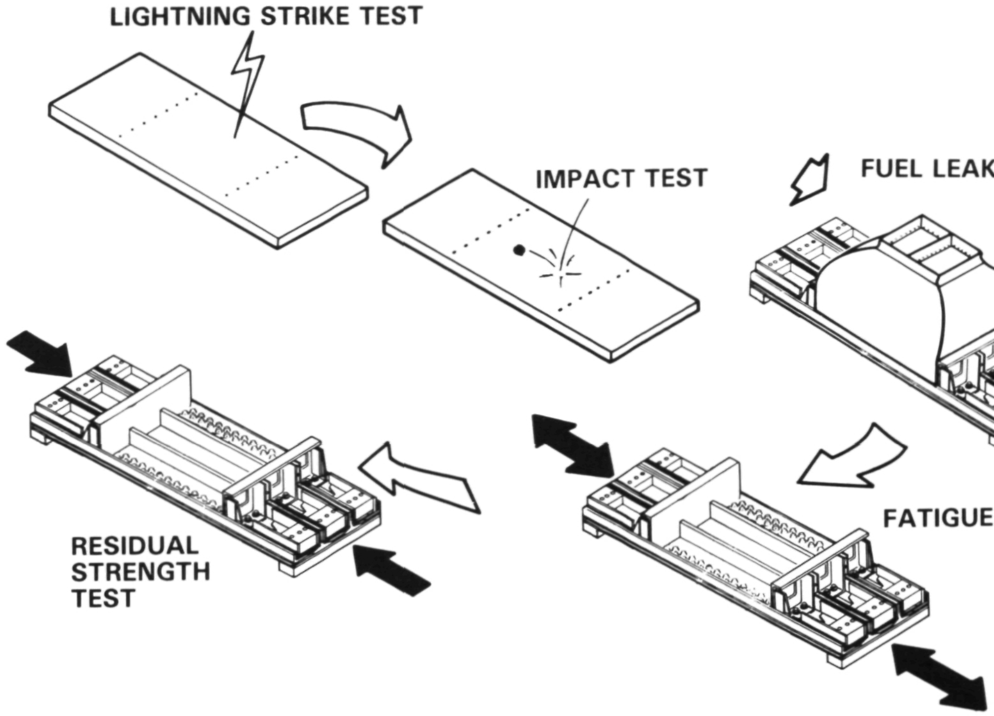


Figure 18. - Technology demonstration article test sequence.

Lightning tests.- The lightning strike test setup used for the series of small panel tests was modified and used for the demonstration article test. In the test, the exterior surface of the demonstration article was struck with a Zone 2 restrike of 100,000 amperes at 50,000 volts along a line of fasteners attaching one of the aluminum ribs to the stiffened panel. A 130-knot stream of air was blown across the panel, simulating the airflow over a transport wing at approach speed. The test setup was the same as that shown in Figure 7. The simulated Zone 2 lightning strike on the exterior of the demonstration article removed some exterior paint but resulted in no burn through or interior sparking. Ultrasonic C-scan of the panel indicated that no structural damage was done to the panel by the strike.

Fuel pressure tests.- To conduct the fuel-pressure tests a fuel enclosure was fabricated and mounted over the demonstration article between the simulated ribs on the back side of the panel. The enclosure was secured by attachments through the ribs and the panel edge-closure angles. Openings were provided for filling and viewing of the fuel simulant, Shell Oil Company "Pella A." Provisions were made to pressurize and drain fuel (Figure 19).

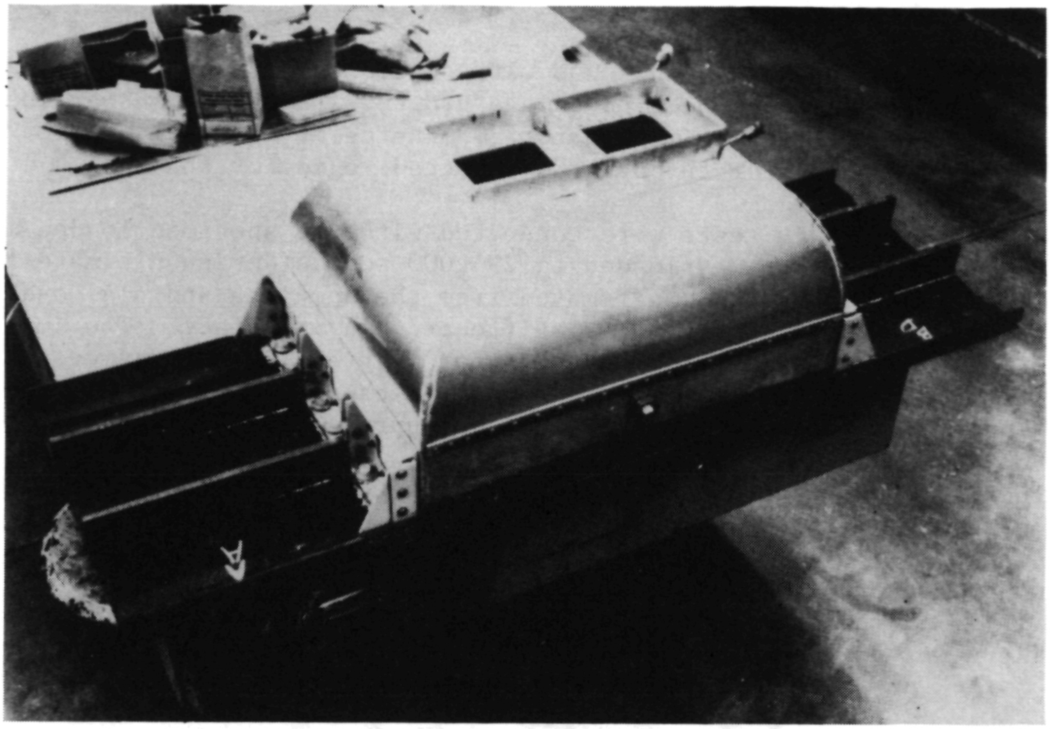


Figure 19. - Fuel enclosure.

After the lightning strike test and ultrasonic inspection, the fuel tank enclosure was attached to the demonstration article and the assembly was proof-tested for leaks at 10 psig. No leaks were observed through the fasteners or through the faying surfaces in graphite-to-graphite joints or graphite-to-aluminum joints.

The enclosure was then removed and the stiffened panel was impacted with 30 ft-lb, using a 0.5-inch-diameter hemispherical steel tup attached to a 12-pound falling weight. The impact produced barely visible damage to the exterior surface of the panel and did not break the Chemglaze paint on the backside of the panel. The damage area as measured by A-scan was 3.64 in^2.

Following the inspection of the 30 ft-lb impact damage on the panel, the fuel tank was reinstalled on the demonstration article. The fuel tank was filled with fuel simulant containing a yellow fluorescent dye, and the tank was pressurized to 10 psig and held at that pressure for 30 minutes. During the test no leaks were observed at the 30 ft-lb impact location or from the fasteners or faying surfaces of the spar-to-stiffened panel.

Structural tests.- The structural tests included fatigue test, pressure test, and then a residual strength test. The fatigue test simulated one lifetime of loading and consisted of 36,000 cycles of compression/tension loads at -98,000 lb/49,000 lb. Every 1000 cycles, a high-load cycle of -157,000 lb/78,500 lb was incorporated. At the completion of the fatigue tests, the impact damaged area of the panel was ultrasonically inspected and it was determined that there had been no damage growth. The specimen was then pressure-tested with fuel simulant for 30 minutes at 15 psig. No leaks occurred at any joints or through the impacted region of the panel.

Residual strength tests were conducted with the specimen loaded in compression. The specimen was loaded to 294,000 pounds(design ultimate load) without failure. The load was removed from the specimen and a second impact damage was put into the panel at a stiffener/skin interface. The impact was accomplished using a spring-loaded impactor with a 1.0-inch-diameter hemispherical tup. The impact energy of 32 ft-lb caused no visual damage. Ultrasonic inspection revealed a damage size of approximately 2.0 square inches.

The panel was then loaded in compression to failure, which occurred at -338,500 pounds (115 percent of design ultimate load). Failure occurred at the location of the first impact damage site. The compressive strains recorded for two gages near the failure location ranged from approximately 5860 to 6370μ in/in (Reference 4). The average axial strain in the panel at the time of failure was 5300μ in/in.

CONCLUSIONS

Compared to aluminum, wing surfaces constructed with graphite composites offer a large weight savings if design-allowable strains can be increased from the current level of approximately 4000μ in/in to 6000μ in/in. Numerous graphite/resin materials were evaluated during this program. Tests on laminates fabricated with high strain-to-failure graphite fibers combined with tougher resins indicate that the desired strain allowable for tension can be obtained. However, for greater post-impact compression strength, significant material improvements are desirable.

Two items were evaluated in the area of fuel containment; sealing of cover-to-substructure joints and fuel leaks through low energy impact damaged laminates. Various solutions to these problems were investigated by coupon and panel tests. An effective solution for fuel sealing was determined to be the conventional fuel tank sealing techniques used for joints in metal structures. This technique uses polysulfide sealant for faying surfaces, fastener topcoat, and fillets. It was also determined that a 0.005-inch thick coating of a flexible polyurethane paint on the inside surface of the wing skin would prevent fuel leaks due to barely visible impact damage.

Swept-stroke lightning strikes to unprotected graphite/epoxy stiffened panels caused internal sparking and a large amount of structural damage. During this program, several surface protection materials and fastener treatments were tested to solve this problem. A reliable and efficient solution was a surface protection material consisting of a graphite/aluminum wire fabric and a fastener treatment of polysulfide and a plastic cap. This combination of materials proved effective in eliminating sparking and reducing structural damage.

The technology developed in this program was verified by the fabrication and test of a full-scale section of a blade-stiffened wing cover including the spar-to-cover and rib-to-cover joints. This structure was subjected to an extensive series of tests including; fuel pressure cycles, a swept-stroke lightning strike, one lifetime of fatigue loads, impact damage, and a residual strength test. The specimen test results exceeded design requirements for all test conditions.

REFERENCES

1. Griffin, C.F., "Fuel Containment and Damage Tolerance in Large Composite Primary Aircraft Structures - Phase I - Summary Report," NASA Contractor Report 166083, March 1983.

2. Sandifer, J.P., "Fuel Containment and Damage Tolerance in Large Composite Primary Aircraft Structures - Phase I - Testing," NASA Contractor Report 166091, March 1983.

3. "Standard Tests for Toughened Resin Composites - Revised Edition," NASA Reference Publication 1092, July 1983.

4. Sandifer, J.P., et al, "Fuel Containment and Damage Tolerance in Large Composite Primary Aircraft Structures - Phase II - Testing," NASA Contractor Report 172519, 1985.

5. Chapman, A.J., "Standard Test Evaluation of Graphite Fiber/Resin Matrix Composite Materials for Improved Toughness," NASA Technical Memorandum 86298, September 1984.

6. Griffin, C.F., et al, "Advanced Composite Aileron for L-1011 Transport Aircraft - Design and Analysis," NASA Contractor Report 165635, April 1981.

7. Jackson, A.C., et al, "Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft, Phase II - Final Report - Design and Analysis" NASA Contractor Report 165634, April 1981.

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		6. Performing Organization Code 76-20
7. Author(s) Charles F. Griffin and Arthur M. James		8. Performing Organization Report No.
		10. Work Unit No.
9. Performing Organization Name and Address Lockheed-California Company P.O. Box 551 Burbank, California 91520		11. Contract or Grant No. NAS1-16856
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15. Supplementary Notes

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16. Abstract

This program addressed the key technology areas of fuel containment, lightning protection and damage tolerance for wing surface panels. The damage-tolerance characteristics of high strain-to-failure graphite fibers and toughened resins were evaluated. Test results show that conventional fuel tank sealing techniques are applicable to composite structures. Also, techniques were developed to prevent fuel leaks due to low-energy impact damage. For wing panels subjected to swept stroke lightning strikes, a surface protection of graphite/aluminum wire fabric and a fastener treatment proved effective in eliminating internal sparking and reducing structural damage. The technology features developed in this program were incorporated and demonstrated in a test panel designed to meet the strength, stiffness and damage tolerance requirements of a large commercial transport aircraft. The panel test results exceeded design requirements for all test conditions. The results of this program provide key technology data for the use of composite materials in transport aircraft wing structure. Wing surfaces constructed with composites offer large weight savings if design allowable strains for compression can be increased from current levels. To obtain the increased strain allowable, improvements in materials are desirable.

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