nasa-CR-166,083

NASA Contractor Report 166083

NASA-CR-166083
19850021720

FUEL CONTAINMENT AND DAMAGE TOLERANCE IN LARGE COMPOSITE PRIMARY AIRCRAFT STRUCTURES

Charles F. Griffin

LOCKHEED-CALIFORNIA COMPANY
Burbank, California 91520

Contract NAS1-16856
March 1983

FOR EARLY DOMESTIC DISSEMINATION

Because of its significant early commercial potential, this information, which has been developed under a U S Government program, is being disseminated within the United States in advance of general publication. This information may be duplicated and used by the recipient with the express limitation that it not be published Release of this information to other domestic parties by the recipient shall be made subject to these limitations

Foreign release may be made only with prior NASA approval and appropriate export licenses This legend shall be marked on any reproduction of this information in whole or in part

Review for general release March 31, 1985

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

FUEL CONTAINMENT AND DAMAGE TOLERANCE IN
LARGE COMPOSITE PRIMARY AIRCRAFT STRUCTURES

CHARLES F. GRIFFIN
Senior Research Specialist
Lockheed-California Company
Burbank, California 91520

ABSTRACT

A program is being 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 are the following: a) the preliminary design of damage tolerant wing surfaces using composite materials, b) the evaluation of fuel sealing and lightning protection methods for a composite material wing, and c) an experimental investigation of the damage tolerant characteristics of toughened resin graphite/epoxy materials.

This paper will present the design concepts investigated for the upper and lower surfaces of a composite wing for a transport aircraft and will discuss the relationship between weight savings and the design allowable strain used within the analysis. The results of experiments to compare the fuel sealing characteristics of bolt-bonded joints and bolted joints sealed with a polysulphide sealant will also be reviewed. Data from lightning strike tests on stiffened and unstiffened graphite/epoxy panels will also be presented. A wide variety of coupon tests were conducted to evalute the relative damage tolerance of toughened resin graphite/epoxies. Data from these tests will be presented and their relevance to the wing surface design concepts discussed.

INTRODUCTION

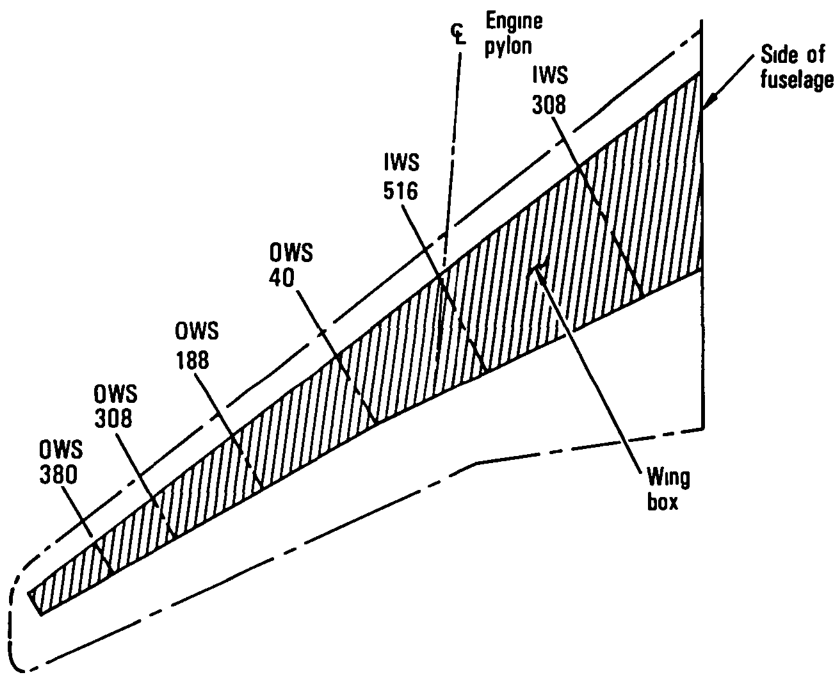
Current applications of composite materials to aircraft structure, most of which are stiffness critical secondary structural components and medium size primary structural components, have demonstrated weight savings from 20 percent to 30 percent. The greatest impact on aircraft performance and cost will be made when these materials are used for fabrication of primary wing and fuselage structures which 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 two-phase program to identify and resolve technical problems associated with fuel containment and damage tolerance of composite material primary wing structure for transport aircraft. The program is sponsored by the National Aeronautics and Space Administration as part of the Aircraft Energy Efficiency (ACEE) Composites Structures Program. This paper presents the results of the first phase of this program.

The first phase of the program included the following activities: preliminary design of composite material wing surfaces for a transport aircraft, evaluation of high strain-to-failure graphite fiber in conjunction with a toughened resin, the investigation of lightning strike behavior of stiffened composite material panels, and the evaluation of fuel sealing methods for bolted joints.

WING SURFACE PRELIMINARY DESIGNS

The baseline wing selected for the study was from the L-1011 transport aircraft. To guide the conceptual design activity a criteria document was compiled which included the wing geometry, loads, stiffness requirements, environmental conditions, and manufacturing constraints. Typical design ultimate axial loading intensities and shear stiffness requirements for the six wing stations investigated are presented on Figure 1. These loads and stiffnesses were used in conjunction with out-of-plane loads such as those due to fuel pressure to do the preliminary sizing of the wing surface structure. Environmental conditions considered for materials selection included: temperature extremes of -65°F to 180°F, resistance to fluids such as fuel, hydraulic fluid and water, and Zone 2 lightning strikes. The damage tolerance criteria stated that for cases where the damage cannot be detected by visual inspection, the structure shall be designed such that the damaged structure can withstand design ultimate loads. For large damage, such as might occur during flight due to uncontained engine failures, the structure must be able to withstand design limit load.



SURFACE \ WING STATION		OWS 380	OWS 308	OWS 188	OWS 40	IWS 516	IWS 308
Upper Surface	Axial load (10^3 lb/in)	−3.7	−6 6	−13 0	−20 4	−25 6	−19.6
	Shear stiffness (10^3 lb/in)	503	581	858	916	932	936
Lower surface	Axial load (10^3 lb/in)	3 6	7.1	13 2	20.7	27.1	15.1
	Shear stiffness (10^3 lb/in)	542	600	1205	1205	2184	1061

Figure 1. Wing Loads and Stiffness Requirements

Wing surface designs were classified into two categories: conventional designs, and damage tolerant designs. For the damage tolerant designs, stiffener geometries and/or skin laminate orientations were analyzed which may offer resistance to delamination propagation or crack growth. A summary of the wing surface designs investigated is presented on Figure 2. Each design was optimized relative to skin thickness and orientation, and stiffener geometry and spacing for the structural criteria and manufacturing constraints. Three design allowable strain levels were used for each design to investigate potential weight savings as a function of design allowable strain.

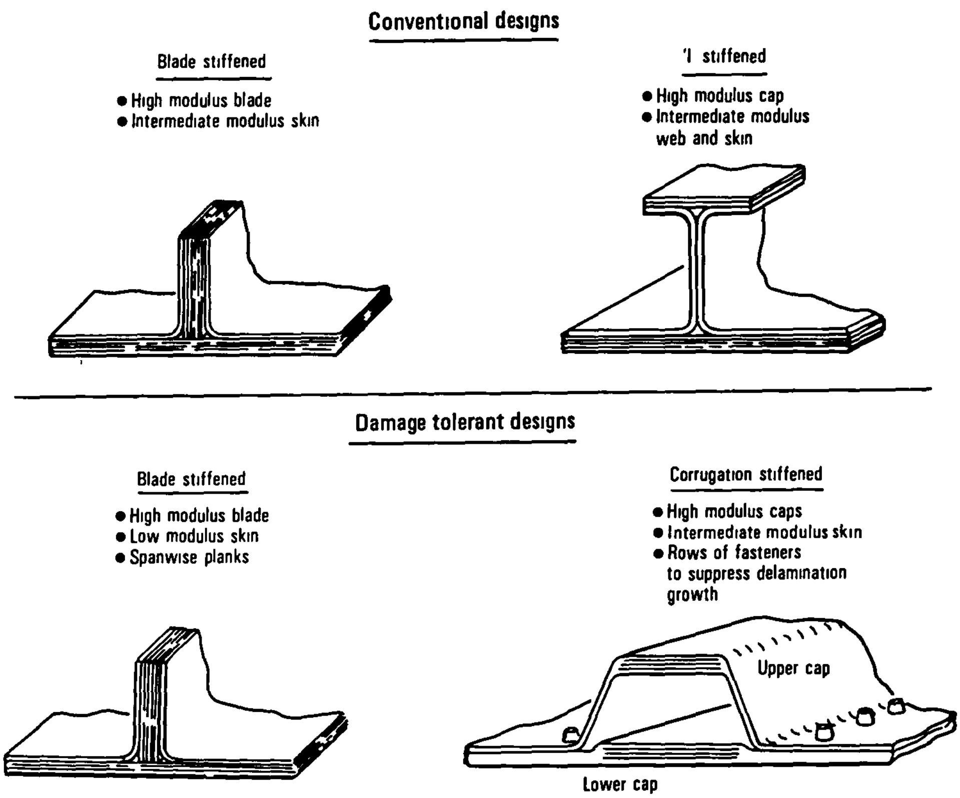


Figure 2. Wing Surface Design Concepts

Results of design trade-offs for conventional designs indicated very little difference in weight savings potential between the blade stiffened design and the 'I' stiffened design. However, the blade stiffened design offers several manufacturing advantages such as less complex tooling and easier attachment to substructure. Upper and lower surface weight savings are displayed for three design allowable strains for the blade stiffened design on Figure 3. Note that the larger weight saving for the lower surface is due to the fact that the metal baseline wing was designed for a lower allowable in tension (45 ksi) than compression (60 ksi). The trends presented on Figure 3 indicate the potential weight savings available if current design strain [1] levels can be increased by improvements in fiber and resin materials and/or design modifications to enhance damage tolerance.

Two design modifications were evaluated which might enhance damage tolerance. The first technique was to redesign the skin orientation of the blade stiffened configuration to reduce the amount of 0-degree direction plies to approximately 10 percent. Avery, Bradley, and King [2] showed that this approach, either as an all

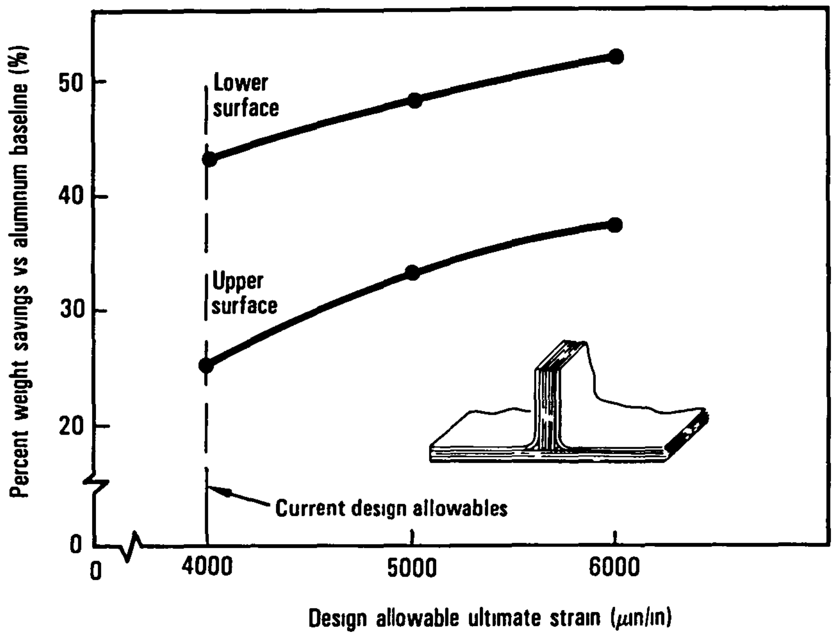


Figure 3. Upper and Lower Surface Weight Savings for Conventional Blade Stiffened Design

graphite laminate or a fiberglass/graphite hybrid, offered a dramatic increase in fracture toughness compared to an all graphite/epoxy quasi-isotropic laminate. The second technique investigated was to limit delamination propagation by using a line of fasteners. A comparison of these designs compared to the optimum blade stiffened design for the upper surface is presented in Table 1. The damage tolerant design concepts have approximately the same weight savings potential as the optimum blade stiffened design, however, in most cases they add manufacturing complexity. It would appear that the most effective way to obtain greater weight savings is to increase design allowable strains by using improved materials.

Table 1. Comparison of Upper Surface Designs

Stiffener Configuration	Skin Configuration	Saved Weight ①	Producibility
Blade	Graphite/epoxy (optimum orientation)	37 8%	Good
Blade	Graphite/epoxy (10% 0^{o}/80% $\pm 45^{o}$/ 10% 90^{o})	36 8%	Fair
Blade	Graphite/epoxy and glass/epoxy hybrid (10% 0^{o} GL/80% $\pm 45^{o}$ GR/ 10% 90^{o} GL) ②	33.1%	Fair
Corrugated	Graphite/epoxy (optimum orientation)	37 1%	Poor

① For strain allowable of 6000 μ in/in ② GR = graphite, GL = glass

INVESTIGATION OF TOUGHENED RESIN GRAPHITE/EPOXY MATERIALS

High strain to failure graphite fibers (1.4 percent tensile elongation) used in conjunction with toughened resins offer a potential to increase design allowable strain levels. To evaluate this potential three graphite/epoxy materials were tested: AS4/3502, a high strain fiber in a standard epoxy resin, and two composites with toughened epoxy resins, AS4/2220-1 and Celion/982.

Quasi-isotropic panels, forty-eight (48) plies thick were fabricated with each material and subjected to impact tests. For these tests a 25 in. x 7 in. portion of the laminate was clamped to a steel plate with a 5 in. by 5 in. opening. The panel was struck in the center of the opening with a 12 lb. impactor which had a 1/2 in. hemispherical diameter hardened steel tip. After impacting, the panels were inspected visually and ultrasonically to ascertain the amount of damage. Figure 4 presents the damage area versus the impact energy for the three materials. Both panels constructed with toughened epoxies had less damage at the lower impact energy levels than did the baseline material. At higher energy levels, where the laminates were being partially punctured by the impactor, the damage area of the baseline material (AS4/3502) was less than the toughened resin materials.

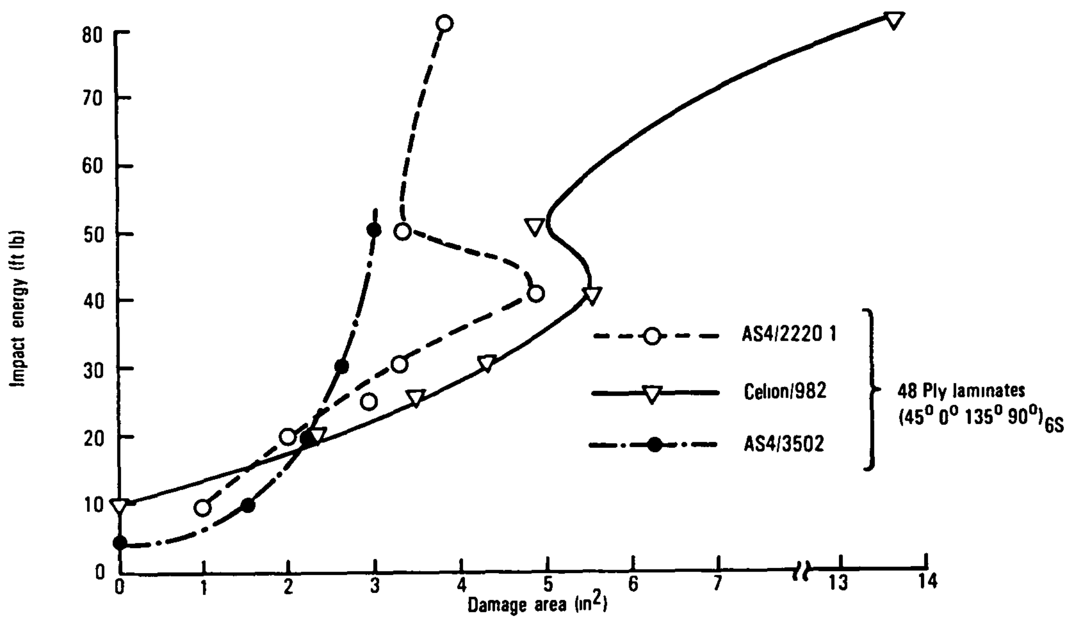


Figure 4. Impact Energy Versus Damage

Tests were then conducted on each material to determine the effect of impact damage on compression strength. The coupon used for these tests was 5 in. wide by 12 in. long. The test fixture simply-supported the coupon at the sides and clamped it at the loaded edges. This technique of stabilizing the coupon allows the out-of-plane deflections associated with delamination growth. A full description of the procedures for this test have been reported by NASA [3]. Each coupon was instrumented with back-to-back axial strain gages located away from the damaged area.

A comparison of the compressive strain to failure of impacted laminates for the baseline material, AS4/3502, and a toughened resin material, AS4/2220-1, is presented on Figure 5. For impact energies at the 20 ft-lb magnitude the toughened resin composite had a failure strain 18 percent greater than the baseline material. This

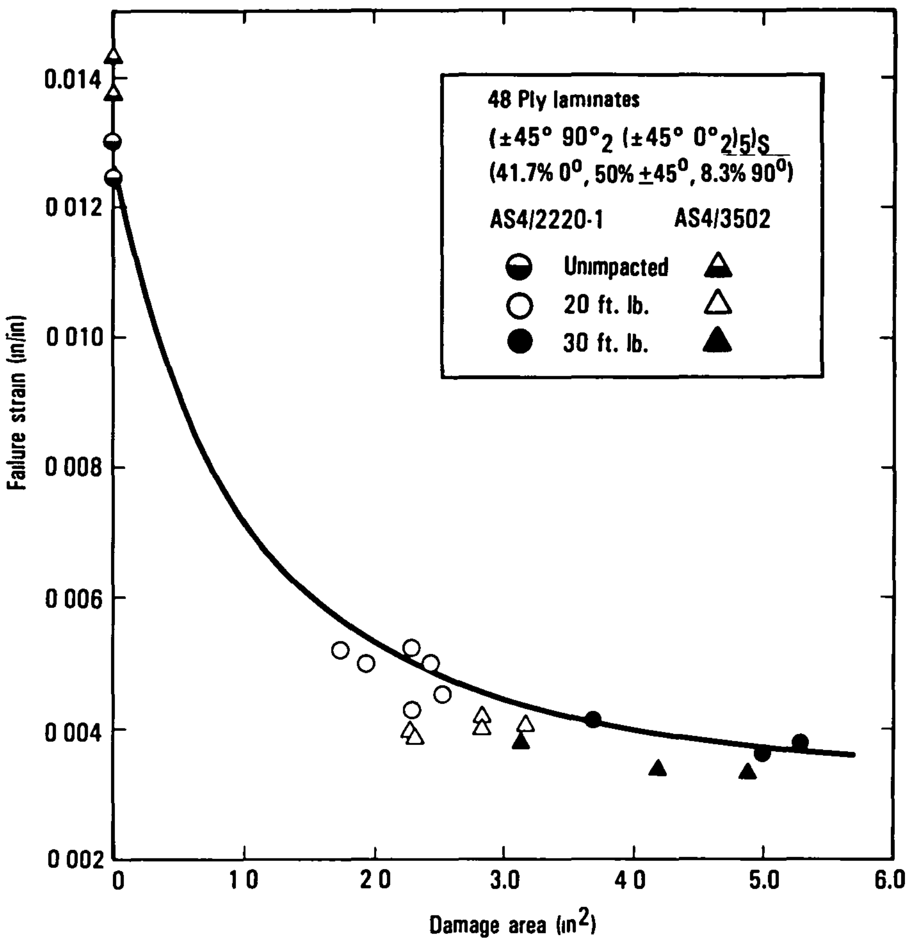


Figure 5. Compression Failure Strain After Impact

improvement decreases to only 9 percent at the 30 ft-lb impact level. These data and the data from comparative tests on AS4/2220-1 and Celion/982, shown on Figure 6, indicate that neither material offers the improvement in impacted compression strain-to-failure needed to substantially increase design allowable compression strains.

The notched tensile strengths of the AS4/3502 and AS4/2220-1 materials were determined by conducting tensile tests on coupons having open holes. Coupons 2.0 in. wide by 14.0 in. long, with either a 0.25 in. diameter hole or 0.50 in. diameter hole were tested. A comparison of the data for these materials, shown on Figure 7, indicates that the composite with the toughened resin, AS4/2220-1, had superior tensile strength for both the unnotched and notched conditions. This difference in tensile strength was also evident in comparative tensile tests on 0-degree laminates where the average failure strains for AS4/3502 and AS4/2220-1 were 0.0116 in/in and 0.01418 in/in, respectively. These data indicate that with a combination of high strain fiber and a toughened resin, a design allowable tensile strain of 0.0060 in/in (0.25 in. diameter notched condition) is attainable. However, many more tests on a wide variety of laminate orientations and environmental conditions must be conducted to establish a tension design allowable for high strain fibers in combination with toughened resins.

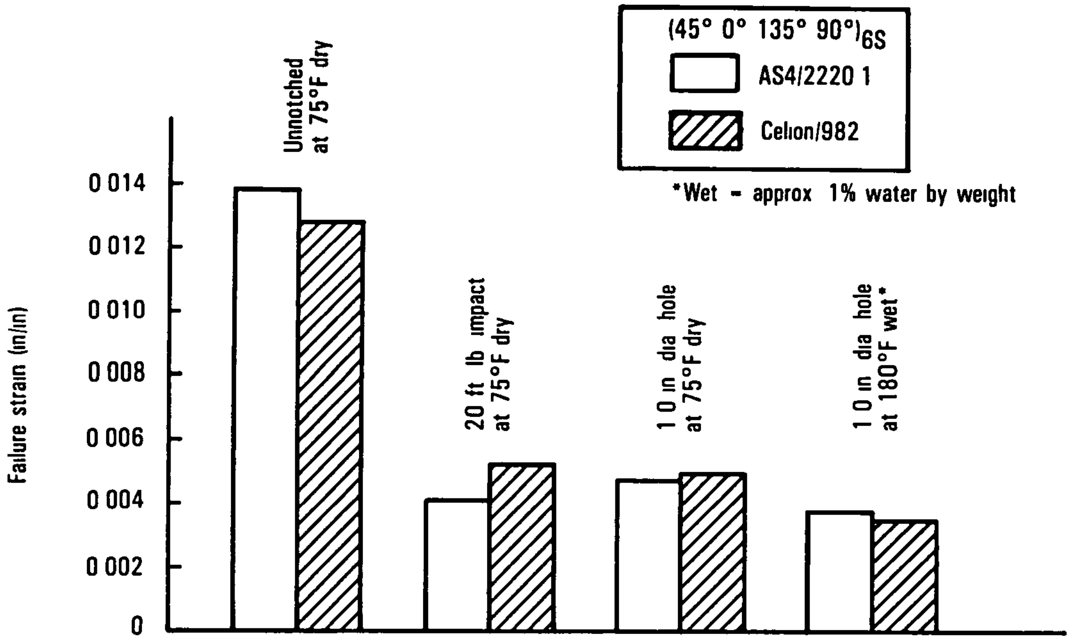


Figure 6. Comparison of Toughened Resin Composites for Compression Loads

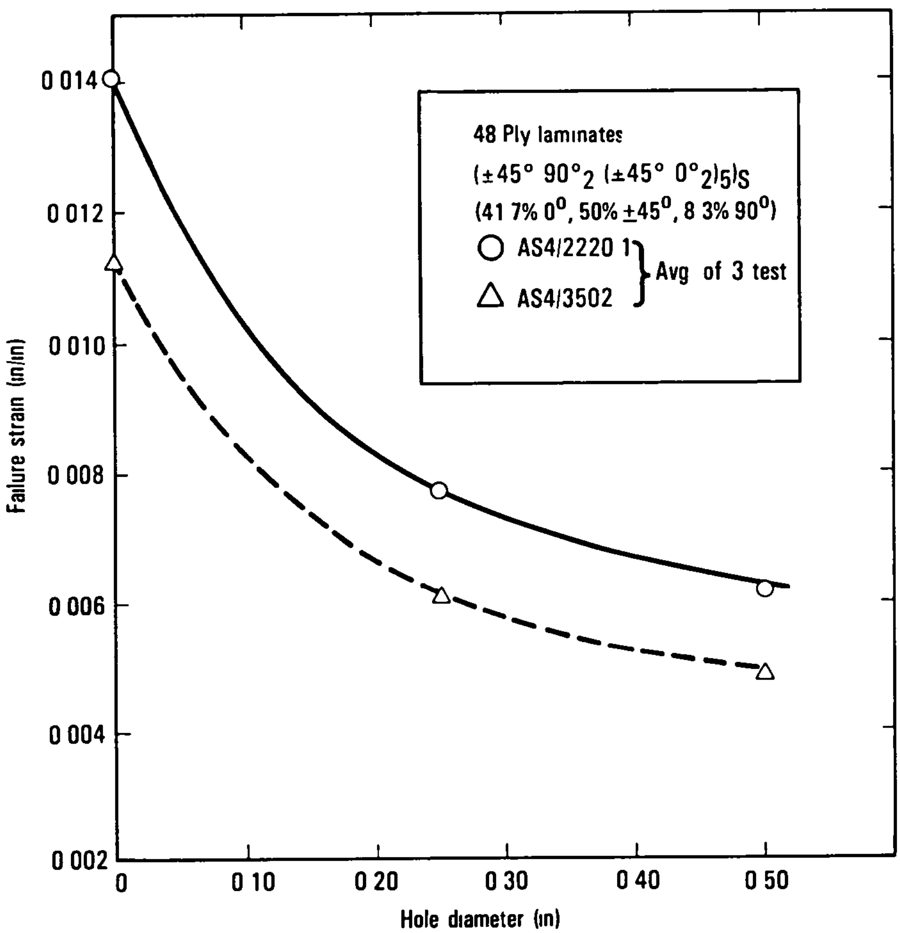


Figure 7. Comparison of AS4/3502 and AS4/2220-1 Laminate Tension Data

FUEL CONTAINMENT

A preliminary assessment was made of the fuel containment capabilities of graphite/epoxy wing structures from two aspects: joint sealing and fuel leakage after impact.

The cover-to-substructure and cover spanwise joints are potential sources of leakage in a wing box which contains fuel. Two methods were evaluated to seal mechanically fastened joints: the conventional approach using a sealant, and an adhesively sealed joint. Comparative tests were conducted using the single lap specimen shown on Figure 8. The specimen was designed to be critical in bolt bearing and had a design ultimate load of 6460 lb. All coupons were constructed of AS4/3502. One-half of the specimens were sealed with a polysulfide sealant and the remainder were bolt-bonded with an AF 10 adhesive. Fillets and fastener collars were sealed with polysulfide sealant on both types of coupons. The fuel simulant used for the tests was Shell Pella A with fluorescent dyes added to enhance visibility with ultraviolet light.

Constant amplitude fatigue tests were conducted on four specimens of each type of joint. For these tests a fuel pressure of 15 psi was applied in combination with 36,000 cycles of axial load at 30 percent of design ultimate load (R = -0.3) and 36 cycles of load at 48 percent of design ultimate load (R = -0.3). No leakage was detected in any of the specimens for this loading.

A second group of four (4) specimens were used to evaluate the effects of sustained axial load in combination with 15 psi fuel pressure. After 300 hours at

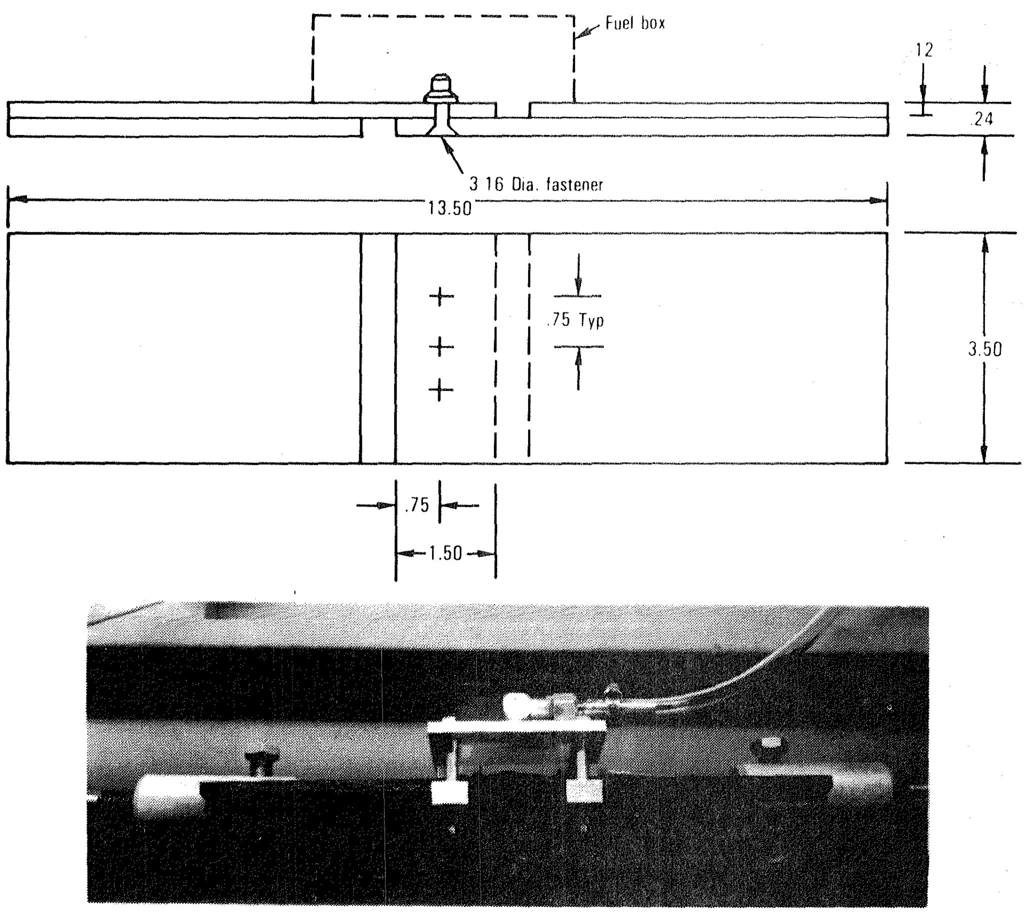


Figure 8. Fuel Sealing Specimen Design

30 percent of design ultimate load and 200 hours at 62 percent of design ultimate load, no leaks had been detected. Another group of four (4) specimens were thermally cycled 100 times between -65°F to 140°F with 15 psi fuel pressure. No leaks occurred during the thermal cycling. All of these specimens were then tested in tension to failure in combination with a fuel pressure of 15 psi. The results of these tests, shown in Table 2, indicate that all specimens withstood design ultimate load before leakage occurred. Typically the failure in the adhesively sealed joints occurred in the adherents, whereas for the polysulfide sealed joints the sealant ruptured causing leakage to occur.

Another potential source of fuel leakage in a wing box constructed with graphite/epoxy is impact damage. A preliminary evaluation of this threat was made using 0.25 in. thick unpainted graphite/epoxy panels impacted at various energy levels and then subjected to fuel pressure on the side opposite to the impact. As shown in Table 3, impacted samples of AS4/2220-1, Celion/982, and AS4/3502 leaked fuel after a very short time at low fuel pressure. In fact, one specimen, impacted at 15 ft-lb leaked within 72 hours with just the full fuel box (approximately 2 in. deep) placed on top of the specimen. Note that at these low impact levels the impact damage was not visually detectable and that neither the front surface nor the back surface of the laminate appeared to be ruptured. During the second phase of this program additional tests will be conducted to evaluate the effect of coatings to prevent fuel leakage after impact.

Table 2. Fuel Sealing Test Data

·Type of Sealant	Fastener Hole Condition	Leak Load (lb) ②	
		Sustained Load Specimens	Thermally Cycled Specimens ①
AF-10	Nominal Dia (.1890 – 1920)	9 448	8 549
AF-10	Maximum Dia (1930 – 1940)	10 700	8 822
Polysulfide	Nominal Dia (1890 – 1920)	6 602	7 236
Polysulfide	Maximum Dia (1930 – 1940)	6 586	7 203

① Tests conducted at -65°F ② Design ultimate load = 6460 lb

Table 3. Impacted Laminate Fuel Leak Test Results

Material ①	Impact Energy (ft-lb)	Delamination Area (in^2)	Fuel Pressure (psi)	Time to Leak (hrs)
AS4/3502	10	1 30	10	1
AS4/3502	10	1 40	5	1
AS4/3502	15	1 60	~0	<72
AS4/2220-1	10	84	10	3.25
AS4/2220-1	20	2 05	5	<22
AS4/2220-1	25	2 89	5	<22
Celion/982	20	1.63	5	8<24
Celion/982	25	3 44	5	8<24

① All laminates 25 in thick

LIGHTNING STRIKE BEHAVIOR

A potential problem with fuel containing wing boxes constructed with graphite/epoxy 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 the engine is a Zone 2 (swept stroke) region.

To evaluate the lightning strike behavior of graphite/epoxy wing skins two unstiffened and six stiffened (see Figure 9) panels were fabricated and tested. These panels had no lightning strike protection and were painted on the outside surface. The stiffeners were attached to the skins with mechanical fasteners. On one-half of the stiffened panels the fastener heads were recessed below the skin surface and the resulting depression was filled with sealant prior to painting to prevent the lightning from attaching to the fasteners.

All of the panels were tested by Lightning and Transients Research Institute for 100,000 ampere swept stroke lightning current levels. 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. The test results, shown in Table 4, indicate that sparking is not a problem with the unstiffened panels even though they sustained a sizeable amount of damage. However, for the stiffened panels sparking occurred in six out of seven tests, irrespective of fastener countersink treatment. In addition to sparking, the stiffened panels also had substantial exterior and internal damage.

Results of these tests indicate that some type of lightning strike protection is required for graphite/epoxy surfaces of fuel containing wings. Additional tests are being conducted to ascertain if the fastener head recess in conjunction with lightning strike protection such as nickel plated graphite fiber fabric on the exterior of the skin will eliminate sparking.

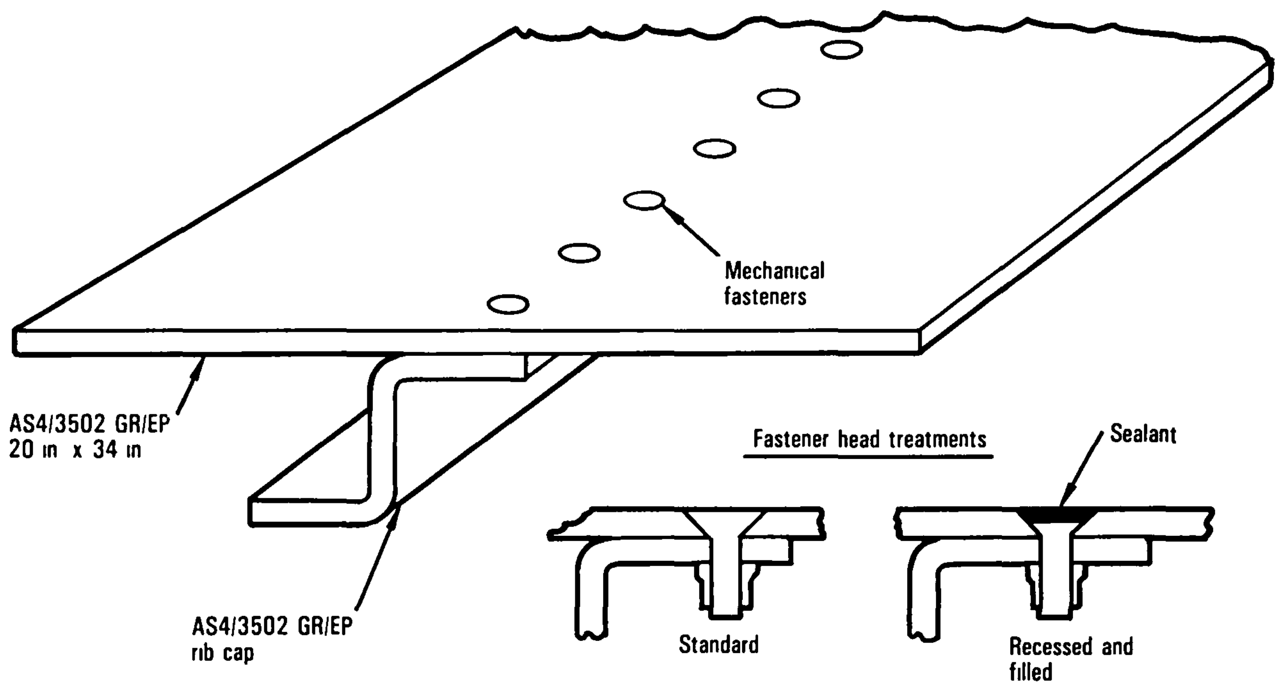


Figure 9. Lightning Strike Test Panel

Table 4. Summary of 100 KA Swept Stroke Tests

Configuration	No. Plies	Countersink Type	Sparking During Test	Visual Front Surface Damage	Visual Back Surface Damage	Damage Area ① (in^2)
Unstiffened	20	–	No	Yes	No	10.5
	20	–	No	Yes	No	13.8
Stiffened	20	Standard	Yes	Yes	Yes	12.5
	20	Recessed	Yes	Yes	No	4.7
	28	Standard	Yes	Yes	No	5.3
	28	Recessed	Yes	Yes	No	13.4
	48	Standard	No	Yes	No	2.2
	48	Standard	Yes	Yes	Yes	15.2
	48	Recessed	Yes	Yes	Yes	16.6

① Measured from ultrasonic inspection

CONCLUSIONS

Preliminary design studies predict 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 combined with currently available tougher resins indicate that the desired strain allowable for tension can be obtained. However, for greater post impact compression strength significant improvements are required.

Based on the data from a limited number of 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. However, the fuel containment capability of a graphite/epoxy tank could be compromised by low energy impact damage. Preliminary tests on impacted 0.25 in. thick graphite/epoxy laminates indicated fuel leakage even though there was no visible damage on either side of the laminate.

Swept stroke lightning strikes to unprotected graphite/epoxy stiffened panels caused internal sparking and a large amount of structural damage. Protection systems must be evaluated to determine their effectiveness to eliminate sparking and reduce the amount of damage.

REFERENCES

1. Griffin, C. F., and Ekvall, J. C., "Design Allowables for T300/5208 Graphite/Epoxy Composite Materials", Journal of Aircraft, v. 19, no. 8, August 1982, 661-68.

2. Avery, J. G., Bradley, S. J., and King, K. M., "Battle Damage Tolerant Wing Structural Development Program Final Report", Naval Air Systems Command Report, March 1979.

3. Anon, "Standard Tests for Toughened Resin Composites", NASA Langley Research Center, NASA RP-1092, May 1982.

1. Report No	2. Government Accession No	3. Recipient's Catalog No
NASA CR-166083		

4. Title and Subtitle
Fuel Containment and Damage Tolerance in Large Composite Primary Aircraft Structures

5. Report Date
March 1983

6. Performing Organization Code
76-23

7. Author(s)
Charles F. Griffin

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

13. Type of Report and Period Covered
Contractor Report
Oct. 1981 to Jan. 1983

12. Sponsoring Agency Name and Address
National Aeronautics and Space Administration
Washington, DC 20546

14. Sponsoring Agency Code

15. Supplementary Notes
a) Langley Technical Monitor: Herbert A. Leybold
b) Presented at 6th Conference on Fibrous Composites in Structural Design, January 1983

16. Abstract

A program is being 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 are the following: a) the preliminary design of damage tolerant wing surface using composite materials, b) the evaluation of fuel sealing and lightning protection methods for a composite material wing, and c) an experimental investigation of the damage tolerant characteristics of toughened resin graphite/epoxy materials.

This paper will present the design concepts investigated for the upper and lower surfaces of a composite wing for a transport aircraft and will discuss the relationship between weight savings and the design allowable strain used within the analysis. The results of experiments to compare the fuel sealing characteristics of bolt-bonded joints and bolted joints sealed with a polysulphide sealant will also be reviewed. Data from lightning strike tests on stiffened and unstiffened graphite/epoxy panels will also be presented. A wide variety of coupon tests were conducted to evaluate the relative damage tolerance of toughened resin graphite/epoxies. Data from these tests will be presented and their relavance to the wing surface design concepts discussed.

17. Key Words (Suggested by Author(s))
Composites, Materials, Primary Aircraft Structure, Damage Tolerance, Graphite/Epoxy, Fuel Containment

18. Distribution Statement
~~FEDD~~ Distribution

Subject Category 24

19. Security Classif (of this report)	20. Security Classif (of this page)	21. No of Pages	22. Price
Unclassified	Unclassified	13	

Available: NASA's Industrial Applications Centers

End of Document