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NASA
Technical Memorandum 4195

AVSCOM
Technical Report 90-B-004

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AUGUST 1990

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Evaluation of Composite Components on the Bell 206L and Sikorsky S-76 Helicopters

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1990

Summary

Progress is described on two programs to evaluate structural composite components in flight service on Bell 206L and Sikorsky S-76 commercial helicopters. Forty ship sets of composite components that include the litter door, baggage door, forward fairing, and vertical fin have been installed on Bell 206L helicopters that are operating in widely different climates. Component installation started in 1981, and selected components are being removed and tested at prescribed intervals over a 10-year evaluation. Four horizontal stabilizers and 11 tail rotor spars that are production components on the S-76 helicopter are being tested after prescribed periods of service to determine the effects of the operating environment on their performance. Concurrent with the flight evaluation, the materials used to fabricate the components are being exposed in ground racks and tested at specified intervals to determine the effects of outdoor environments. In this paper, results achieved from 123 000 flight hours of accumulated service on the Bell 206L components and 53 000 flight hours on the Sikorsky S-76 components are reported. Seventy-eight Bell 206L components have been removed and tested statically. Results are presented after 7 years of ground exposure of materials used to fabricate the Bell 206L components. Results of tests on 4 Sikorsky S-76 horizontal stabilizers and 11 tail rotor spars are presented. Panels of material used to fabricate the Sikorsky S-76 components that were exposed for 6 years have been tested and results are presented.

Introduction

Over the past 15 years (yr), NASA has sponsored programs to build a data base and establish confidence in the long-term durability of advanced composite materials in transport aircraft structures (ref. 1). Primary and secondary components have been installed on commercial aircraft to obtain worldwide flight service experience. Flight environments for transport aircraft and helicopters are quite different, and the behavior of composite components in the two environments may differ substantially.

As a result, in 1978 NASA and the U.S. Army Research and Technology Activity (AVSCOM) initiated the first major program to evaluate helicopter composite components in flight service. A contract was awarded to design, fabricate, certificate, and install 40 ship sets of composite litter doors, baggage doors, forward fairings, and vertical fins on Bell 206L helicopters (manufactured by Bell Helicopter Textron, Inc.). The specific objective of this program is to determine the long-term durability of composite airframe structures in the operational environment of light commercial helicopters. Such helicopters often operate for extended periods in remote areas with primitive maintenance facilities and near unimproved areas where damage from tree limbs, rocks, sand, and other debris is commonly encountered.

In 1979 NASA and the U.S. Army Research and Technology Activity initiated a second research program to determine the residual strength of composite helicopter components after specified periods of flight service. A contract was awarded to Sikorsky Aircraft Company to evaluate the flight service performance of 4 horizontal stabilizers and 11 tail rotor paddles on Sikorsky S-76 helicopters and to determine the residual strength of each composite component after removal from service. The composite components are production parts for the S-76.

The S-76 composite components were chosen to compare real-time in-service environmental effects with accelerated laboratory test results and analytical predictions for both static and dynamic loaded primary structures. The tail rotor is designed primarily for cyclic fatigue loading, whereas the horizontal stabilizer is designed for static loading. Realistic environmental factors established through flight service and residual strength testing of these components will allow a more efficient design of composite components for future helicopters.

Concurrent with the two flight service programs, materials used to fabricate the components are being exposed in ground racks at seven sites and are tested at prescribed intervals to determine the effects of outdoor environments.

This paper describes the design, flight service experience, and postservice testing of each composite component and the ground-based exposure of material specimens. The residual strength of components after flight service and the strength of specimens after outdoor exposure are reported and compared with baseline values.

The Bell 206L Program

Component Description

A total of 45 ship sets of litter doors, baggage doors, forward fairings, and vertical fins were manufactured for the Bell 206L helicopter (fig. 1). A detailed description of the design, fabrication, and certification of each component is reported in reference 2. A brief description of each component follows.

Litter door. The litter door is located on the left side of the aircraft as shown in figure 1. The door is 26 in. wide by 46 in. high. Schematics of the litter door are shown in figure 2. The door consists of outer and inner skins of Kevlar-49[1] fabric/F-185[2] epoxy composite material. Each skin contains areas that are reinforced with unidirectional Kevlar-49/F-560[2] epoxy composite material. Each skin was fabricated separately and then the two skins were secondarily bonded together to form the door. A plexiglass-type window was bonded directly to the door with EC3549[3] adhesive. The weight of the metal door is 13.10 lbm, whereas the weight of the composite door is 8.20 lbm for a weight saving of 37.4 percent. Bell Helicopter was responsible for the design and fabrication of the litter door.

The design of the litter door was controlled primarily by two loading conditions required for Federal Aviation Administration (FAA) certification: (1) an outward aerodynamic load that includes the reaction loads from the hinge points of the cabin door, and (2) the weight of the litter door and cabin door plus a 50-lbf downward force at the cabin door handle. The latter loading condition simulates a person pulling on the cabin door when both doors are open.

Baggage door. The baggage door is also located on the left side of the aircraft as shown in figure 1. The door is 37.5 in. long by 23.4 in. wide. A photograph of the baggage door is shown in figure 3. The door consists of Kevlar-49 fabric/LRF-277[4] epoxy composite material face sheets bonded on 3.1 lbm/ft^3 Nomex[5] honeycomb core. Areas around the hinges and latches were reinforced with additional plies of Kevlar-49 fabric/LRF-277 epoxy. Weights of both the composite and metal baggage door are 2.90 lbm. The baggage door offered no weight savings but remained in the program to access the effects of long-term durability. The Brunswick Corporation was responsible for the design and fabrication of the baggage door.

The design of the baggage door was based primarily on two loading conditions required for FAA certification: an outward aerodynamic load and a downward load caused by pulling on the door latch in the open position.

Forward fairing. Location of the forward fairing on the aircraft is shown in figure 1. The fairing is 35.9 in. long, 29.0 in. wide, and 13.0 in. high at the aft end. A photograph of the fairing is shown in figure 4. Most of the fairing consists of single-ply Kevlar-49 fabric/CE-306[6] epoxy composite material skin that was co-cured on a 4.5 lbm/ft^3 Klegecell[7] foam core. Areas around the hinges and latches were reinforced with additional plies of Kevlar-49/CE-306 epoxy. The weight of the metal forward fairing is 8.60 lbm, whereas the composite fairing weighs 7.26 lbm for a 15.6-percent weight saving. Bell Helicopter was responsible for the design and fabrication of the forward fairing.

Design and certification tests of the fairing were based on an outward aerodynamic pressure load.

Vertical fin. The vertical fin is used for directional stability in forward flight and is located on the aircraft as shown in figure 1. A photograph of the fin is shown in figure 5. The fin is 79.0 in. high and the chord varies between 12.0 in. and 19.0 in. The vertical fin is a full-depth sandwich structure with T-300[8]/E-788[9] graphite/epoxy composite material face sheets on a Fibertruss[10] core. Fibertruss is a high-strength, high-stiffness fiberglass core. An aluminum-alloy screen was bonded to the exterior surface of the skin to provide lightning protection. The leading edge of the vertical fin is a 2-ply Kevlar-49 fabric/epoxy skin attached to the structural box. The tail skid is a tapered filament-wound S-glass[11]/epoxy tube with a short length of steel tubing and standard abrasion pad attached at the tip. The weight of a metal vertical fin is 15.30 lbm and the weight of the composite vertical fin is 12.30 lbm, which results in a 19.6-percent weight saving. Bell Helicopter was responsible for the design and fabrication of the vertical fin.

Design and certification of the vertical fin was based on three design conditions, two static and one fatigue loading. The first static test condition simulated aerodynamic loading only. The second static test condition simulated an aircraft landing in the tail-down attitude. Fatigue tests were conducted on specimens that simulate the fin-to-fuselage attachment structure of the fin. The fatigue tests were conducted at room temperature after the specimen had been conditioned at 120°F and 95-percent relative humidity for 42 days (1000 hr).

Ground Exposure Specimens

Concurrent with the flight service program, material test specimens are being exposed at five locations

[1] Kevlar-49: Registered trademark of E. I. du Pont de Nemours & Co., Inc.

[2] F-185; F-560: Manufactured by Hexcel Corporation.

[3] EC3549: Manufactured by 3M Company.

[4] LRF-277: Manufactured by Brunswick Corporation.

[5] Nomex: Trademark of E. I. du Pont de Nemours & Co., Inc.

[6] CE-306: Manufactured by Ferro Corporation.

[7] Klegecell: Manufactured by American Klegecell Corporation.

[8] T-300: Manufactured by Amoco Performance Products, Inc.

[9] E-788: Manufactured by U.S. Polymetric Company.

[10] Fibertruss: Manufactured by Hexcel Corporation.

[11] S-glass: Manufactured by Ferro Corporation.

on the North American Continent (fig. 6). The selected locations are in the general areas where the composite components are being flown. Each location contains one rack as shown in figure 7. The racks were installed in 1980 and contain five trays, each for removal after 1, 3, 5, 7, and 10 years of exposure. A tray contains 24 each of tension, short-beam-shear (SBS), and Illinois Institute of Technology Research Institute (IITRI) compression specimens and four 2.0-in-wide specimens to provide information on the weathering characteristics of each material system. The tension, compression, and SBS specimens are painted with a polyurethane paint (IMIRON[12]) that is used on the flight service helicopters.

The four composite material systems in the ground exposure program are given as follows: (1) Kevlar-49 fabric (style 281)/F-185 epoxy $[0/45/0]_s$; (2) Kevlar-49 fabric (style 120)/LRF-277 epoxy $[0/90/\pm45]_s$; (3) Kevlar-49 fabric (style 281)/CE-306 $[0/90]_s$; and (4) T-300/E-788 $[0/\pm45/0]_s$ graphite/epoxy. These material systems correspond to those used for the litter door, baggage door, forward fairing, and vertical fin, respectively.

Flight Service Evaluation

A total of 40 ship sets of composite components have been supplied to operators as kits for installation on aircraft that are located in the four geographical areas shown in figure 8. The areas selected include: a hot, humid, salt-spray environment (U.S. Gulf Coast); a cold rocky environment (Alaska); a cold, damp, pollution-prone environment (East Canada and Northeast United States); and a hot, dry environment (Southwest United States). Each component is inspected annually or after 1200 hr of service for evidence of damage, repair, excessive wear, or weathering. At the conclusion of the first, third, fifth, seventh, and tenth year of flight service, selected components are removed and returned to Bell Helicopter for static testing. Prior to testing, each component receives the same nondestructive inspection that was required during manufacturing. Test results are compared with design strength requirements.

Evaluation of Ground Exposure Specimens

A tray of ground exposure specimens is removed at a prescribed interval of time, sealed in a plastic bag, and shipped to the Langley Research Center. The tray remains in the sealed bag until static testing is initiated. All tests are performed at room temperature on six replicates for each specimen type. The tests are performed in accordance with the following ASTM standards: (1) Tension-D3039, (2) SBS-D2344, and (3) Compression-D3410 using the IITRI test fixture.

Specimens used to characterize moisture absorption were cut from the tested tension specimens. A 0.5-in-long section was cut from the undamaged area of the tension specimen as soon as possible after completion of testing. The paint was removed by sanding, but using caution not to remove an excessive amount of the outer ply. Each specimen was weighed after the paint removal. A 0.5-in-long specimen was also removed from the unpainted exposure specimen and weighed prior to being used for moisture content determination. All specimens were stored in sealed plastic bags between the different operations.

All specimens used to characterize moisture absorption were placed in a vacuum oven at 140°F. Each specimen was weighed periodically to determine weight loss as a function of time.

Results and Discussion

Flight service components. Installation of the composite components on the Bell 206L began in March 1981. Aircraft flying with these components have accumulated 122 355 hr through the end of 1988. The aircraft with the highest flight time has flown 9606 hr. Over one-half (67 919 hr) of the total time has been accumulated by aircraft flying in the Gulf of Mexico area. Next in flying time is Northeast United States and East Canada with 38 195 hr followed by Alaska with 8321 flight hours. Aircraft in Southwest United States have flown 7920 hr. As of April 1989, 51 components were flying, 78 components were removed for testing, 9 components were being reinstalled, 15 components were lost because of crashes or were damaged beyond repair, and 11 components could not be located.

The litter door has had very few problems with the composite material skins. However, a major problem occurred with the metal hinges that are used to hold the cabin door. The hinges were underdesigned and failed in service when someone pushed the cabin door open too far. New high-strength hinges have been installed and the litter doors are back in service.

Buckling occurred in the outer skin of the litter door on four helicopters parked in the Southwest United States desert during the summer. The probable cause of buckling is the thermal mismatch between the Kevlar-49/epoxy skins and the plexiglass window. This window was bonded to the exterior skin. The coefficient of thermal expansion of the plexiglass window is 4.5×10^{-6} in/in/°F, and the coefficient of thermal expansion for the Kevlar/epoxy

[12] IMIRON: Trademark of E. I. du Pont de Nemours & Co., Inc.

skin is near zero. Personnel at this desert facility have taken surface-temperature measurements on aluminum helicopter structures with the same external paint scheme as the composite skins, and their results show that the temperatures typically reach 200°F to 225°F during the summer. The bond between the door structure and window was broken near the buckle on the four doors. The doors were modified and a rubber seal was bonded between the door structure and window to permit relative thermal expansion. Other normal service problems have occurred such as broken windows, bumps, and scratches, all of which were repaired at the operator's repair facilities according to repair instructions received with the kits.

Of the four components, the baggage door has the poorest service record. All baggage doors have been removed from service because of large, unrepairable voids between the outer skin and the Nomex core. A destructive inspection of doors that were removed from service indicated very little filleting between the outer skin and core, thus resulting in a poor bond between the outer skin and core. During manufacture, the outer skin was co-cured with the Nomex core but the inner skin had an adhesive layer between the skin and the core.

Another service problem with the baggage door is cracking of the unsupported corners. This corner cracking of the baggage doors is caused by people accidentally hitting the corners of the door with baggage and other gear to be stowed in the baggage compartment. This is an aesthetic problem rather than a functional problem but it does distract from the appearance of a commercial vehicle. The door and some of the adjacent structure would have to be redesigned to eliminate this corner cracking.

The forward fairing has had the fewest service problems. Until the 1985 inspection, the only service-related problem was associated with the use of the fairing as an antenna base. Field operators use the flat upper surface of the fairing to mount their communications antennas and they had to bond a metal plate to the underside of the fairing for grounding. The 1985 field inspection revealed that two helicopters operating in the Gulf Coast area had developed cracks on the inside surface near each latch. Both aircraft have been in service since 1981 and have flown 4193 hr and 5409 hr, respectively. The fairings were repaired with fiberglass, according to maintenance instructions, at the operator's repair facility.

The graphite/epoxy vertical fin has an excellent service record. Its only problem has been the cracking of paint on the 2-ply Kevlar unsupported leading edge. This cracked paint was caused by field personnel using the fin as a handhold in ground handling. Two fins have been struck by lightning and one fin was repaired and returned to service. The other fin was returned to Bell for analysis and residual strength testing.

One of the severest effects of the Gulf Coast environment on metallic components is corrosion. Operators typically start to repair corrosion on metal fins after $1\frac{1}{2}$ to 2 years of service. By 6 years in service, the leading edge, trailing edge, and several other parts of the metal fins have been rebuilt. The graphite fins on aircraft flying in the Gulf Coast area have been in service for up to 7 years without a single maintenance problem related to corrosion.

As part of the flight service program, selected components are removed from service and tested to the same simulated aerodynamic loading as that used in FAA certification. Seventy-eight components have been removed for testing. The exposure region, exposure time, flight hours, and postservice failure loads for each component are given in tables 1–4. The exposure times range from 12 to 84 months (mo) and the flight times range from 386 to 6750 hr.

Failure loads (table 1) for 15 litter doors vary from 901 to 1768 lb. These failure loads are the total loads on the door including the hinge reaction from the cabin door. Failure loads as a function of flight hours are shown in figure 9. Each open symbol represents the failure load for a different exposure area. Also shown in figure 9 are two strength requirements, ultimate strength and design strength. The ultimate strength is the usual strength that an aircraft component must meet or exceed at all times. The design strength shown is the strength that an unconditioned component (as-fabricated) must meet or exceed. The design strength is intentionally greater than the ultimate strength and is the product of the ultimate strength and the environmental factor determined from the environmentally conditioned material coupon specimens (ref. 2). This strength is shown for reference only since the components tested are between as-fabricated and fully conditioned. The solid symbol represents the average baseline failure load (reported in ref. 2), which is the average of five components selected at random from the production lot of 45 components. The range of failure loads for the baseline tests is also shown in the figure. All litter doors had failure loads that exceeded the ultimate strength requirement of 635 lb (ref. 2), and nine of the doors had failure loads that exceeded the design strength of 1229 lb. It is acceptable for the failure load of doors that have been environmentally exposed to be below the design strength as long as the load is higher than the ultimate strength. Failure loads do not appear to be a function of exposure time. Initially in the postservice testing (ref. 3), only the failure load was to be determined and compared with the baseline and certification loads. During testing of the third set of components the recording of deflections to limit the load was started. From these load-deflection data for each component a stiffness could be determined for each component. This stiffness for each component gives another indication of composite material response to environmental effects since some of the failure loads are determined by metal hinge failures or latch pins slipping from the test fixtures. The stiffness of the litter door as a function of flight time is shown in figure 10. The stiffness is calculated using measurements for the deflection at midspan of the small post (see section C-C in fig. 2) as the load was applied. The stiffness shown for certification is the average of three tests used for certification. A large scatter in the measured stiffness is shown in figure 10. This large scatter is acceptable when considering that the door is installed in a fixture that simulates the aircraft attach points and is loaded to limit load with water bags to simulate the uniform aerodynamic loading. A difference of only 0.025-in. deflection at limit load will change the stiffness by a factor of 2. It is unfortunate that the deflections for the baseline tests were not recorded for comparison with components from the same production run. The doors that have been tested have accumulated a total of 23 087 hr of flight service.

Failure loads (table 2) for 26 baggage doors vary from 0.32 to 1.57 psi, and service times for the doors vary from 386 to 6750 hours. Failure loads as a function of flight hours are shown in figure 11. Each open symbol represents the failure load for a different exposure area. The solid symbol represents the average baseline strength. These baggage doors are the components that developed large disbonds and have been removed from service. Most of the test points represent baggage doors that have some disbond between the core and the outer skin. Twenty-one of the doors had failure loads that were below the design strength of 0.70 psi, and nine doors had failure loads that were below the ultimate strength requirement of 0.50 psi (ref. 2). The stiffness of the baggage door as a function of the flight hours is shown in figure 12. The stiffness was determined by measuring the deflection at the center of the door as the loads were applied. The disbonds do not appear to affect the stiffness as much as the strength. The baggage doors that have been tested have accumulated a total of 51 798 hr of flight service.

Failure loads (table 3) for 15 forward fairings vary from 1.80 to 3.93 psi, and service times for these components vary from 386 to 6750 hr. Failure loads as a function of flight hours are shown in figure 13. Each open symbol represents the failure load for a different exposure area. Failure loads vary from 1.80 to 3.93 psi and exceed the design strength of 0.49 psi (ref. 2) by more than a factor of 3. Failure loads for forward fairings tested to determine the baseline strength (ref. 2) varied from 2.2 to 3.4 psi. Failure loads for 8 of the 15 forward fairings tested are between 2.2 and 3.4 psi. The large scatter in the failure load could be the result of variations in the fabrication process. Considerable variation could result from putting down the single ply of Kevlar fabric (style 281) on a compound contoured surface. The lay-up requires cutting and overlapping the fabric at many places to prevent wrinkling on the surface. The stiffness of the forward fairing as a function of the flight hours is shown in figure 14. The stiffness was calculated using measurements for the deflection of the upper surface at a point 14.5 in. from the aft end as the load was applied. The stiffness of all fairings, except one, exceeded the stiffness of the certification fairings. The fairings that have been tested accumulated a total of 23 730 hr of flight time.

Failure loads (table 4) for 15 vertical fins vary from 1.12 to 1.80 psi, and service times for these fins vary from 385 to 6750 hr. Failure loads as a function of flight hours are shown in figure 15. All fins exceed the design strength of 1.05 psi. Failure loads for 12 of the 15 fins are between 1.35 and 1.57 psi, which is the range of failure loads for the baseline tests (ref. 2). One fin that was struck by lightning is identified in table 4. The fin was damaged at the top with no damage in the structural box. This fin failed at 1.23 psi with no apparent effect from the lightning strike. The environment does not appear to affect the failure load of the vertical fins. The stiffness of the vertical fin as a function of flight time is shown in figure 16. Stiffness was calculated using measurements for the tip deflection as the simulated aerodynamic load was applied. The certification deflection data for the fin are not available. The vertical fins that have been tested have accumulated 26 139 flight hours.

Ground exposure specimens. In the summer of 1985 the exposure racks (fig. 7) located at Cameron, Louisiana, and on the offshore oil platform were destroyed by hurricanes. Therefore, all the following data for 5 and 7 years of exposure are from the three remaining sites: Hampton, Virginia; Toronto, Canada; and Fort Greely, Alaska.

The baseline strengths for the as-fabricated ground exposure specimens are given in table 5. Each table entry is the mean strength of the six replicates tested. The residual compressive and short-beam shear strengths of exposed painted specimens are shown in figures 17 and 18, respectively. Each point shown in the figures for 1 year or 3 years of exposure is the mean of 30 tests (5 racks and 6 replicates of each material), whereas each point for 5 years or 7 years of exposure is the mean of 18 tests (3 racks and 6 replicates of each material). The mean strength results shown in each figure are normalized by the mean baseline strength. Scatter bands in the baseline strength are also shown in figures 17 and 18. The residual compression strength of exposed painted specimens shown in figure 17 varies between 88 and 101 percent of baseline. Kevlar-49/LRF-277 material has the lowest strength retention of 88 to 90 percent of baseline. For the first 5 years of exposure the other materials exceeded the lower baseline scatter band of 96 percent. At 7 years of exposure Kevlar-49/F-185 and T-300/E-788 materials retained 93 percent of the baseline strength. The short-beam shear strength of exposed painted specimens (fig. 18) varies between 89 and 104 percent of baseline. Like the compression strength results, the Kevlar-49/LRF-277 material has retained the lowest short-beam shear strength of 89 to 92 percent. The T-300/E-788 material retained the highest strength of 100 to 104 percent of baseline. All materials, except Kevlar-49/LRF-277, exceeded the baseline scatter minimum of 93 percent. The residual tension strength of the exposed specimens after exposure equals or exceeds the baseline strength.

A summary of moisture absorption data for each material as a fraction of composite specimen weight for painted specimens that were exposed for 3, 5, and 7 years is shown in figures 19–22. Each symbol represents a different exposure location, and the solid symbols represent the unpainted specimens. Each data point for painted specimens is the average of six replicates. Summaries of the moisture absorption data for the unpainted specimens are also shown in figures 19–22. Each data point for the unpainted specimens is from a single specimen. Moisture absorption data for the Kevlar-49/CE-306 material are shown in figure 19. No trend is evident after 3 years of exposure. After 5 and 7 years of exposure the painted specimens absorb 0.15 to 0.38 percent (average) more moisture than the unpainted specimens. The painted Kevlar-49/CE-306 specimens appear to be reaching an equilibrium condition of 2.1 percent moisture absorption.

Moisture absorption data for the Kevlar-49/F-185 material are shown in figure 20. This figure indicates that painted specimens absorb more moisture than the unpainted specimens. Actually, the painted specimens absorb up to 0.63 percent (average) more moisture than the unpainted specimens. Moisture absorption data for the Kevlar-49/LRF-277 material are shown in figure 21. The trend in this material is the opposite from the other two Kevlar materials in this program. For the Kevlar-49/LRF-277 material the unpainted specimens absorb more moisture than the painted specimens. By the seventh year of exposure the unpainted specimens are approaching an equilibrium condition of 2.3 percent moisture absorption.

Moisture absorption data for the T-300/E-788 graphite/epoxy material are shown in figure 22. The 0.74-percent absorption for a 3-year exposure at the Gulf of Mexico does appear high. There is no method to determine if this high moisture absorption would continue in the Gulf of Mexico since the rack was destroyed before the removal of specimens after 5 years of exposure. No trend is evident after 3 years of exposure. After 5 and 7 years of exposure the painted specimens absorbed approximately 0.10 percent more moisture than the unpainted specimens, the same trend followed by two of the Kevlar/epoxy systems. Kevlar-49/epoxy materials absorb four to five times more moisture than graphite/epoxy materials because the Kevlar fibers absorb moisture. The average values, for each material, compare well with published values for other Kevlar/epoxy and graphite/epoxy systems (ref. 1).

Sikorsky S-76 Program

Component Description

A total of fifteen S-76 composite components were used in this evaluation: 4 horizontal stabilizers and 11 tail rotor spars. The location on the S-76 of the horizontal stabilizer and tail rotor paddles, which contain the tail rotor spars, is shown in figure 23. A detailed description is given in reference 4 and a brief description of each component follows.

Horizontal stabilizer. A sketch of the left half of the horizontal stabilizer is shown in figure 24. The stabilizer is a full-depth sandwich structure with cross-plied Kevlar-49 fabric/5143[13] epoxy composite material skins and Nomex honeycomb core. The torque tube that joins the left and right sides of the stabilizer is full-depth aluminum honeycomb construction with unidirectional AS1[14] graphite/6350[15] epoxy composite material in the spar caps. The torque tube is overwrapped with cross-plies of Kevlar-49 fabric/5143 epoxy to provide additional torsional rigidity. The composite horizontal stabilizer weighs 40.0 lbm.

The design of the stabilizer was controlled primarily by static load requirements. All production

[13] 5143: Manufactured by American Cyanamid.

[14] AS1: Manufactured by Hercules, Inc.

[15] 6350: Manufactured by Ciba-Geigy.

parts are proof load tested at room temperature prior to installation. For proof load testing the stabilizer is supported at ±25.0 in. from the centerline and a 2400-lbf downward load is applied at the centerline. The deflection of the torque tube is measured and recorded. FAA certification and baseline strengths were achieved by supporting the stabilizer at the aircraft attachment points and applying a load through pads bonded to the stabilizer skin at ±40.0 in. from the centerline. The design limit loads (DLL's) for static tests and the baseline loads for fatigue testing are shown in figure 25. Static tests are performed with the structure at 160°F. Fatigue tests are performed at room temperature.

Tail rotor spar. The tail rotor spar is a solid laminate constructed with AS1 graphite/6350 epoxy composite material. The spar is 52.9 in. long by 3.5 in. wide and is shown in figure 26. The weight of the spar is 14.6 lbm. Two glass/epoxy blades are attached to the spar to form the tail rotor paddle as shown in figure 27.

The tail rotor spar was designed to withstand a high number of cyclic loads. The tail rotor was fatigue tested using the edgewise moment, flatwise moment, torsion, and centrifugal loads illustrated in figure 28. The centrifugal load is kept constant and represents the centrifugal force for a rotor operating at 110 percent of the normal rotor speed. The cyclic loadings are in phase and are held in the same proportions as the absolute values are varied to produce a fatigue fracture in the range from 10^5 to 5×10^6 cycles.

Material Allowables

Using the projected aircraft usage, Sikorsky predicted (ref. 4) the saturation moisture levels in Kevlar-49/5143 to be 2.2 percent and in AS1/6350 to be 1.1 percent. To expedite the development of design allowables for the S-76 program, Sikorsky used accelerated conditioning on coupon specimens for determining material properties. All conditioning was conducted at 87 percent relative humidity and 190°F.

Ground Exposure Panels

Panels of AS1/6350 and Kevlar-49/5143 are being subjected to outdoor ground-based exposure at Stratford, Connecticut, and West Palm Beach (WPB), Florida. The Kevlar-49/epoxy panels are 5 plies thick and the graphite/epoxy panels are 6, 14, and 33 plies thick. Each year, three panels of each material and thickness combination are removed for evaluation. The panel sizes are 8.0 in. by 22.0 in., 6.0 in. by 8.0 in., and 2.0 in. by 6.0 in. The 2.0 in. by 6.0 in. panels were left unpainted for determining the effects of weathering on bare composites, and the other panels were painted with a polyurethane aircraft paint. The 6-ply graphite/epoxy panels are machined into compression and flexure specimens. The 14- and 33-ply graphite/epoxy panels are machined into compression, short-beam shear static, flexure, and short-beam shear fatigue specimens. The 5-ply Kevlar-49/epoxy panels are machined into tension specimens. All exposed specimens are tested at room temperature, and the test data are compared with baseline data for room-temperature dry specimens. The moisture content is determined by cutting the 6.0 in. by 8.0 in. panel into four specimens that are dried at 150°F.

Flight Service Evaluation

Four horizontal stabilizers and 11 tail rotor spars have been removed from aircraft in service over an 8-year period. Since these components are production parts, they receive the normal maintenance inspection for surface damage every 100 flight hours and an inspection for structural damage annually or after 1000 flight hours. One of the stabilizers has been static tested and the remaining stabilizers have been fatigue tested.

Seven of the tail rotor spars have been fatigue tested and the remaining four spars have been cut into short-beam shear specimens that have been subjected to the following three tests: (1) room temperature static, (2) 170°F static, and (3) room-temperature fatigue. After full-scale component testing, coupons have been removed from the components to determine their moisture content.

Results and Discussion

Flight service components. The horizontal stabilizers and tail rotor spars were removed from aircraft operating in the Gulf Coast region of Louisiana. The components have accumulated a combined total of 53 146 hr, 15 496 hr for the stabilizers and 37 650 hr for the tail rotor spars. All components scheduled for testing have been removed from service. The flight hours and exposure times at removal are given in table 6.

Horizontal stabilizers: The first horizontal stabilizer (serial no. 00076; see table 6) removed from service had accumulated 1600 flight hours over a 17-month period. Prior to full-scale testing, the stabilizer was proof load tested in accordance with the procedure required for production acceptance. The proof load deflection for this stabilizer was the same as the corresponding deflection for the stabilizer used for the initial acceptance test. The flight service stabilizer was static tested to failure. Data for strain as a function of percent limit load are shown in figure 29. The tension strain response was linear up to 160 percent of the design limit load (DLL) and then increased at reduced slope until the maximum applied load of 220 percent of the DLL was reached. The compression strain response was linear to 120 percent of the DLL and then increased at a reduced slope until 170 percent of the DLL was reached. The compression strain did not increase after 170 percent of the DLL. At 220 percent of the DLL the stabilizer made a loud "snap" and the load dropped to 150 percent of the DLL. Attempts to increase the load beyond 150 percent of the DLL resulted in an increased deflection until the test fixture limit was reached. Visual inspection of the stabilizer indicated a buckle in the splice plate on the left-hand leading edge of the torque box. A teardown of the component revealed a loss of shear transfer capabilities between the composite material and the metal honeycomb. The stabilizer tested for certification did not fail but reached the maximum deflection allowed by the fixture at 268 percent of DLL.

The second stabilizer (serial no. 00009) was subjected to proof loading and fatigue loading after 56 months of exposure and 3999 hr of flight time. The deflection for this stabilizer at proof load was the same as the corresponding deflection for the stabilizer used for the acceptance test. The same loads were applied for fatigue testing as were used for certification (fig. 25). When no fracture occurred after 5×10^5 cycles, the test was stopped and the loads were increased by 5 percent. At 3×10^5 additional cycles at the higher loads, a fracture occurred in the torque box.

The third stabilizer (serial no. 00021) was subjected to proof loading, static loading, and fatigue loading after 66 months and 4051 flight hours. Visual inspection and coin tapping revealed two small disbond areas in the torque box. One disbond measured approximately 0.75 in. long by 1.50 in. wide and was located at left buttline 3 (3 in. to left of aircraft centerline). The other disbond measured approximately 1.0 in. long by 3.0 in. wide and was located at right buttline 3. An acceptable deflection was measured during proof load testing and indicated no loss of stiffness. The stabilizer was loaded to the design limit load (at 160°F) followed by fatigue testing at room temperature. Because of an error in loading, the applied fatigue loads were 23 percent higher than the baseline loads. After 59 980 cycles, fracture occurred in the torque box. The fractured area was located between right and left buttline 3.

The fourth stabilizer (serial no. 00027) was subjected to proof loading and fatigue loading after 91 months and 5846 flight hours. The deflection for this stabilizer at proof load was the same as the corresponding deflection for the stabilizer used for the acceptance test. The same loads were applied for fatigue as were used for certification (fig. 25). After 437 340 cycles, fracture occurred in the torque box.

Tail rotor spars: Eleven tail rotor spars have been removed for testing. No defects were found during inspections of the spars. Seven of the spars were fatigue tested and the remaining spars were used to obtain specimens for coupon tests. A summary of data for the tail rotor spars is given in table 7 along with data from spars labeled serial numbers 00046 and 00064 (ref. 5). These two spars were removed from a Sikorsky flight test aircraft that was located at West Palm Beach, Florida. The points designated by "First" in table 7 represent the first fracture on one side of the spar. On spars that did not have a complete failure, testing was continued on the other side until fracture occurred, and those results are designated by "Last." Cyclic shear stress as a function of cycles to crack initiation is shown in figure 30 along with the strength of dry spars tested at room temperature for FAA certification. These results indicate a 94-percent strength retention for the exposed spars when compared with strength data from certification tests.

A predicted moisture-time profile (ref. 6) for the tail rotor spars (ref. 5) operating in the Louisiana Gulf Coast region is shown in figure 31. Weather data from Lake Charles, Louisiana, were used in predicting the absorbed moisture. Measured moisture values (table 7) are shown in figure 31 on a plot of percent moisture as a function of exposure time. As can be seen in figure 31, the measured moisture values are below the predicted moisture values for the low exposure time (30–40 months), whereas the measured moisture values are for an exposure time of 70 months higher than the predicted values.

Ground exposure specimens. A summary of the moisture absorption results for exposed panels with 2–6 years of exposure is presented in table 8. These results indicate that the 6-ply AS1/6350 panels exposed at West Palm Beach, Florida, have exceeded the predicted saturation levels of 1.1 percent moisture, whereas the panels exposed at Stratford, Connecticut, have reached the predicted levels. The 14-ply and 33-ply panels are not expected to reach saturation for several years. A predicted moisture-time profile (ref. 6) for the 6-ply panels exposed at Stratford is shown in figure 32. The measured moisture values for each panel (table 8), which are also shown in figure 32, show good agreement with the predicted values. Plots of residual strength as a function of moisture content for flexure, short-beam shear, and tension specimens are shown in figures 33, 34, and 35, respectively. Each individual data point is the mean strength of 18 tests. The solid line in each figure is the residual strength after the accelerated conditioning of the specimens (ref. 4). Residual flexure strengths in figure 33 for exposed 6-ply AS1/6350 laminates exceed 95 percent of the baseline strength and also meet (5 years at West Palm Beach, Florida) or exceed the strength of the accelerated conditioned specimens. Residual short-beam shear strengths in figure 34 for exposed AS1/6350 laminates vary between 72 and 89 percent of baseline and are within 1 percent of the strength of the accelerated conditioned specimens. Residual tension strengths in figure 35 of the Kevlar-49/5143 material vary from 99 to 107 percent of the baseline strength and exceed the strength of the accelerated conditioned specimens by 11 to 24 percent. This material follows the same trend as the tension strength of Kevlar-49 fabric materials used in the Bell 206L program.

Concluding Remarks

Bell 206L Program

Aircraft flying the Bell 206L components have accumulated approximately 123 000 hr of service. The high-time aircraft has accumulated over 9000 hr. Over one-half of the total flight hours have been accumulated in the Gulf of Mexico area. The use of composite material eliminates the metal corrosion problem that is significant on aircraft that fly in the Gulf of Mexico area. The baggage door has the poorest service record because of poor bonding between the exterior skin and the core. This bond could not be repaired, and thus the baggage doors were removed from flight service. The other components have had problems from ground handling or underdesigned metal parts. In general, the composite components have performed well in flight service. Individual components that have been tested have accumulated up to 6750 flight hours and 69 months of exposure time. With the exception of the baggage door, postservice strengths exceed the ultimate strength.

Residual short-beam shear and compression strengths for ground exposure specimens after 5 years of exposure exceeded 88 percent of baseline strengths. The Kevlar-49/LRF-277 material had the lowest retained strength of 88 to 92 percent of baseline strength after exposure. The residual strength of all other materials exceeds 93 percent of the baseline strength after exposure. The tension strength of all specimens after exposure equaled or exceeded the baseline strength.

The Kevlar-49/CE-306 and Kevlar-49/F-185 materials absorb up to 0.6 percent more moisture when painted. The Kevlar/epoxy materials appear to be approaching an equilibrium condition at 2.1 to 2.3 percent moisture content. The T-300/E-788 painted material absorbs approximately 0.1 percent more moisture than the similar unpainted materials.

Sikorsky S-76 Program

The 15 components evaluated in the Sikorsky S-76 program have accumulated over 53 000 flight hours of service. The four horizontal stabilizers removed from service passed the proof load test. A horizontal stabilizer with 17 months of exposure failed at 220 percent of design ultimate load. Two horizontal stabilizers with 56 and 66 months of service have been fatigue tested and failed at 300 000 and 59 980 cycles, respectively, at applied loads exceeding the loads used for FAA certification.

The tail rotor spars retained 94 percent of the baseline fatigue strength after 8 years of exposure. The predicted moisture content for the spars is high for short exposure time (30–40 months) but low for long exposure time (over 70 months).

The six-ply AS1/6350 panels exposed at West Palm Beach, Florida, have exceeded the predicted design moisture level of 1.1 percent, whereas the panels at Stratford, Connecticut, have reached the predicted level. Residual compression and short-beam shear strengths of AS1/6350 exceed 72 percent of the baseline strength after 6 years of exposure. The residual tension strength of Kevlar-49/5143 exceeded the baseline strength. The residual strength after outdoor exposure equals or exceeds the strengths from laboratory-conditioned specimens for all material systems.

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June 6, 1990

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Table 1. History of Litter Doors Removed for Testing

Exposure region	Serial number	Start of service, mo/yr	End of service, mo/yr	Time, mo	Flight time, hr	Failure load, lb
Gulf of Mexico	45373	6/81	5/82	12	879	1009
	45378	2/82	11/84	34	3387	988
	45367	10/81	1/87	64	6750	1618
	ML-A83	(a)	(a)	(a)	(a)	1262
S.W. USA	45614	5/82	10/83	17	386	901
	45418	5/82	1/85	32	1235	1768
	45607	5/82	11/87	58	1992	1592
	45609	5/82	8/83	16	802	1644
N.E. USA and Canada	45141	5/81	7/82	15	870	980
	45028	4/82	11/84	32	1160	1302
	45101	8/81	6/83	23	1413	1115
	45017	3/82	11/84	33	902	1750
	45085	4/82	11/84	32	1369	1492
Alaska	45115	5/82	10/84	29	668	931
	45109	7/84	3/88	45	1284	1643

[a]Unknown.

Table 2. History of Baggage Doors Removed for Testing

Exposure region	Serial number	Start of service, mo/yr	End of service, mo/yr	Time, mo	Flight time, hr	Failure load, psi
Gulf of Mexico	45373	6/81	5/82	12	879	0.91
	45378	2/82	11/84	34	3387	1.39
	(a)	(a)	(a)	(a)	(a)	.60
	45378	2/82	4/87	29	2500	.43
	45524	3/82	3/87	60	4828	.69
	45449	11/82	3/87	53	5837	.46
	45546	8/82	4/88	69	6027	.71
	45367	10/81	1/87	64	6750	.60
S.W. USA	45614	5/82	10/83	17	386	0.46
	45418	5/82	1/85	32	1235	.49
	45607	4/83	12/87	66	1992	.55
	45608	5/82	11/87	64	1317	.49
N.E. USA and Canada	45141	5/81	7/82	14	870	0.50
	45028	4/82	11/84	32	1160	1.57
	45101	8/81	6/83	22	1413	.32
	45017	3/82	11/84	26	902	1.37
	46607	4/83	12/87	56	1824	.50
	45083	6/82	1/88	67	2195	.57
	45085	4/82	11/84	33	1369	.54
	ML-112	(a)	(a)	(a)	(a)	.55
	ML-13	(a)	(a)	(a)	(a)	.37
Alaska	45115	5/82	10/84	29	668	1.39
	45108	12/81	3/88	76	2004	.43
	45109	7/84	3/88	45	1284	.60
	45113	11/81	1/87	63	1772	.54
	45114	3/84	3/88	48	1199	.54

[a]Unknown.

Table 3. History of Forward Fairings Removed for Testing

Exposure region	Serial number	Start of service, mo/yr	End of service, mo/yr	Time, mo	Flight time, hr	Failure load, psi
Gulf of Mexico	45373	6/81	5/82	12	879	1.80
	45378	2/82	11/84	34	3387	2.34
	45367	10/81	1/87	64	6750	2.70
	45535	(a)	(a)	20	500	2.11
	ML-03	(a)	(a)	(a)	(a)	3.68
S.W. USA	45614	5/82	10/83	17	386	2.80
	45418	5/82	1/85	32	1235	3.68
	45607	5/82	11/87	58	1992	3.73
N.E. USA and Canada	45141	5/81	7/82	15	870	2.50
	45028	4/82	11/84	32	1160	2.47
	45101	8/81	6/83	23	1413	1.89
	45017	3/82	11/84	33	902	2.46
	45085	4/82	11/84	32	1369	3.44
Alaska	45115	5/82	10/84	29	668	2.69
	45113	11/81	11/88	84	2219	3.93

[a]Unknown.

Table 4. History of Vertical Fins Removed for Testing

Exposure region	Serial number	Start of service, mo/yr	End of service, mo/yr	Time, mo	Flight time, hr	Failure load, psi
Gulf of Mexico	45378	2/82	4/87	34	3387	1.12
	45373	6/81	5/82	12	879	1.80
	(a)	(a)	(a)	(a)	(a)	1.48
	45367	10/81	1/87	63	6750	1.44
S.W. USA	45608	5/82	11/87	66	2213	1.57
	45614	5/82	10/83	18	386	1.41
	45607	5/82	11/87	66	1992	1.49
	45418	5/82	1/85	32	385	1.50
N.E. USA and Canada	45028	4/82	11/87	32	1160	1.37
	[b]45450	9/81	7/84	36	2661	1.23
	45141	5/81	7/82	15	870	1.60
	45101	8/81	6/83	23	1413	1.51
	45085	4/82	11/84	32	1369	1.49
	45017	3/82	11/84	33	902	1.39
Alaska	45113	1/81	1/87	63	1772	1.49

[a]Unknown.
[b]Struck by lightning.

Table 5. Strengths of As-Fabricated Ground Exposure Specimens

Material	Strength, ksi		
	SBS (a)	Compression	Tension
Kevlar-49/F-185	6.0	20.2	57.4
Kevlar-49/LRF-277	3.9	22.4	83.7
Kevlar-49/CE-306	5.3	18.3	61.1
T-300/E-788	11.2	126.3	126.5

[a]Short-beam shear.

Table 6. Flight Times of S-76 Components Removed From Service

Component	Serial number	Flight time, hr	Exposure time, mo
Horizontal stabilizer	00076	1600	17
	00009	3999	56
	00021	4051	66
	00027	5846	91
Tail rotor spars	00094	2390	29
	00283	1884	37
	00150	2385	38
	00237	2128	42
	00172	2533	39
	00114	3358	52
	00178	3753	51
	00069	4940	69
	00415	5138	68
	00493	5858	97
	00480	5816	100

Table 7. Summary of Data for S-76 Tail Rotor Spars

Serial number	Exposure time, mo	Flight time, hr	Cyclic shear stress, psi (*a*)	Cycles to crack imitation	Moisture content, percent
00046	[b,c]25	150	(First) 3980 (Last) 3980	0.25×10^6 0.38×10^6	0.29
00064	[b,c]25	150	(First) 4320 (Last) 4320	0.35×10^6 0.71×10^6	.32
00094	[d]29	2390	(First) 3890 (Last) 3920	0.286×10^6 0.174×10^6	.26
00283	[d]37	1884		Coupon tests	.36
00150	[d]38	2385		Coupon tests	.40
00237	[d]42	2128	4520	0.267×10^6	.47
00172	[d]39	2533	4270	0.218×10^6	.49
00114	[d]52	3358	4416	0.839×10^6	.56
00178	[d]51	3753		Coupon tests	.60
00069	[d]69	4940	3820	0.146×10^6	.66
00415	[d]68	5138		Coupon tests	.78
00493	[d]97	5858		Coupon tests	(*e*)
00480	[d]100	5816	4640	0.140×10^6	(*e*)

[a]First and last (final) failures.
[b]Reference 5.
[c]In-service location: West Palm Beach, Florida.
[d]In-service location: Gulf Coast Region, Louisiana.
[e]Desorption in progress.

Table 8. Summary of Moisture Content for Exposed Panels

Material	Number of plies	Exposure location (a)	Exposure time, mo	Moisture (by weight), percent
AS1/6350 graphite/epoxy	6	WPB	26	1.02
			35	1.23
			48.5	1.15
			60.5	1.40
			72.5	1.34
		↓	84	(b)
		Stratford	25	.86
			36	1.00
			49	.99
			62	1.13
			73	1.07
	↓	↓	85	(b)
AS1/6350 graphite/epoxy	14	Stratford	25	0.37
			34.5	.48
			48	.44
			61	.64
			72	[c].57
	↓	↓	84.5	(b)
AS1/6350 graphite/epoxy	33	WPB	26	0.27
			35	.37
			48.5	.35
			60.5	.42
			72.5	[c].44
		↓	84	(b)
		Stratford	25	.18
			36	.22
			49.5	.24
			62	.30
			73.5	[c].25
	↓	↓	85	(b)
285/5143 Kevlar/epoxy	5	WPB	26	1.56
			35	2.08
			48.5	1.90
			60.5	1.88
			72.5	2.02
			84	(b)
		Stratford	26	1.53
			37	1.72
			50	1.75
			63	1.92
			74	1.70
	↓	↓	85.5	(b)

[a]Locations: West Palm Beach, Florida, and Stratford, Connecticut.
[b]Desorption in progress.
[c]Estimated.

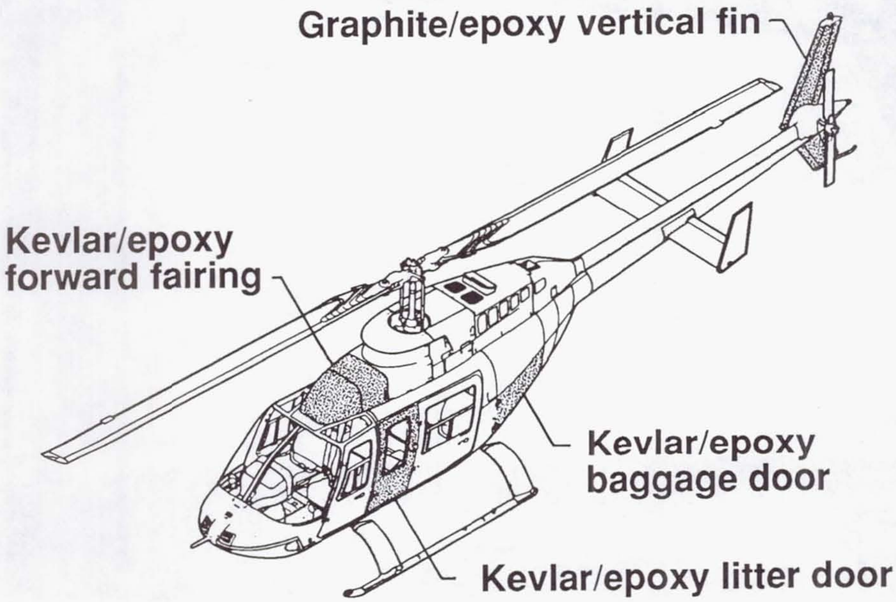


Figure 1. Composite components in flight service on Bell 206L helicopters.

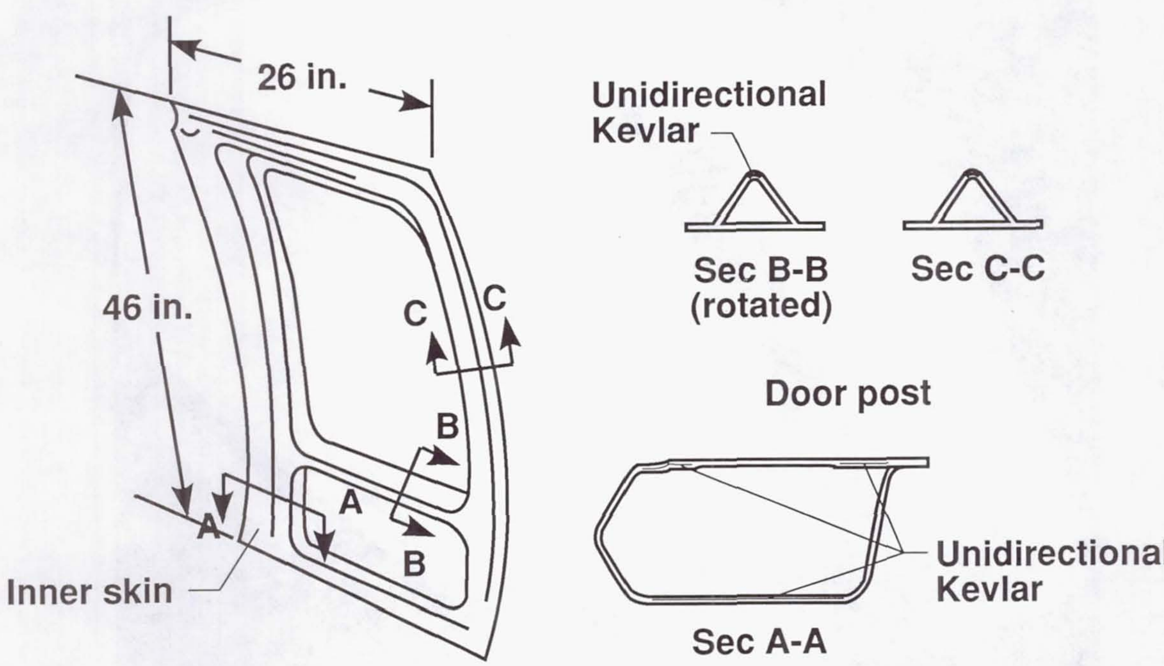
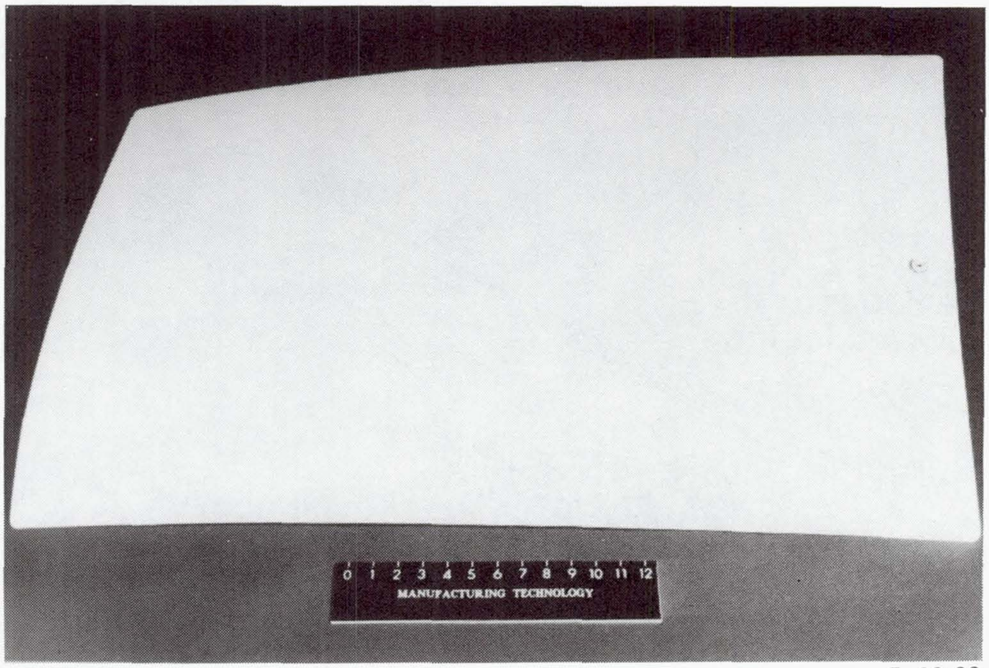
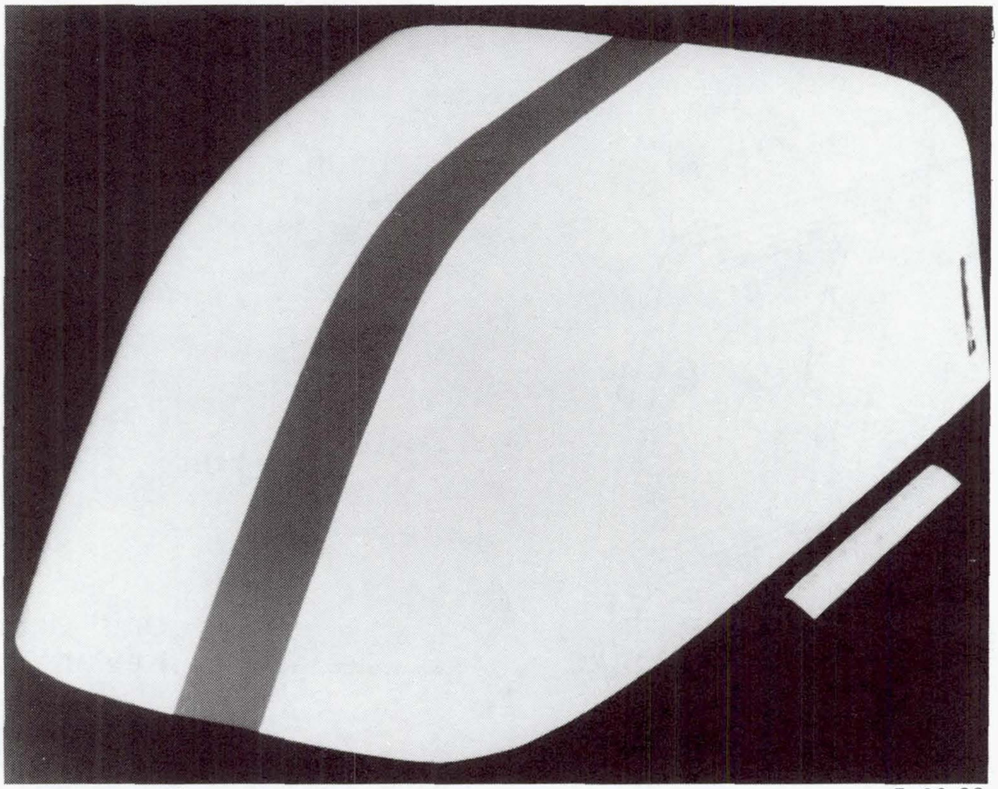


Figure 2. Bell 206L Kevlar/epoxy litter door.



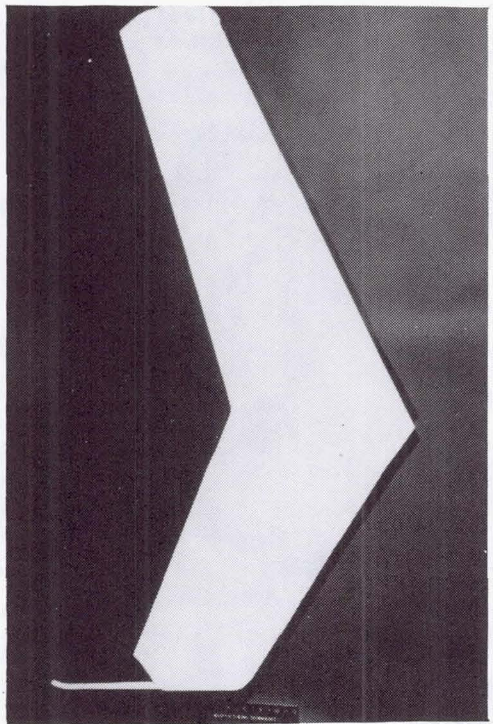
L-90-32

Figure 3. Bell 206L Kevlar/epoxy baggage door.



L-90-33

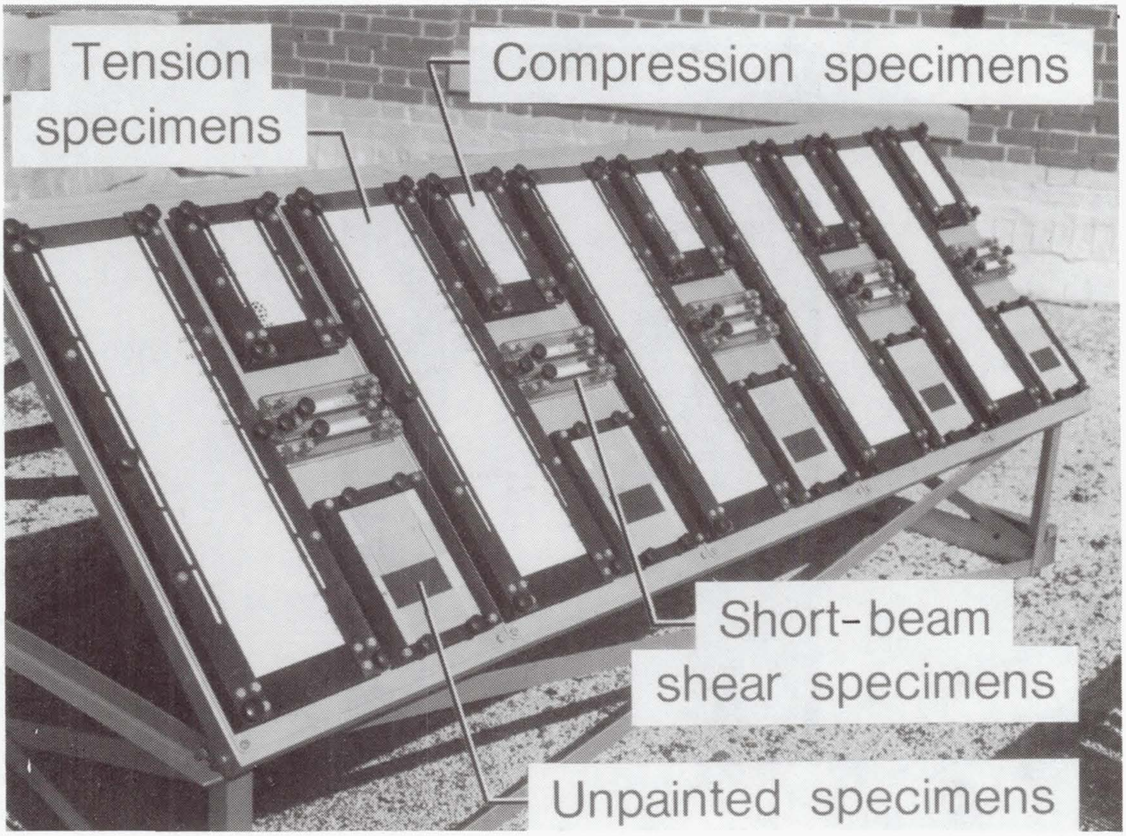
Figure 4. Bell 206L Kevlar/epoxy forward fairing.



L-90-34

Figure 5. Bell 206L graphite/epoxy vertical fin.

Figure 6. Location of environmental specimen exposure racks.



L-90-35

Figure 7. Environmental exposure rack with specimens installed.

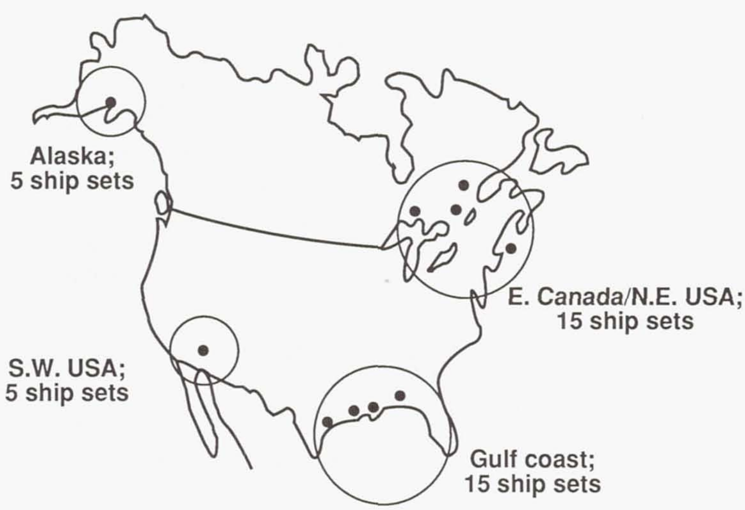


Figure 8. Distribution of Bell 206L helicopters with composite components.

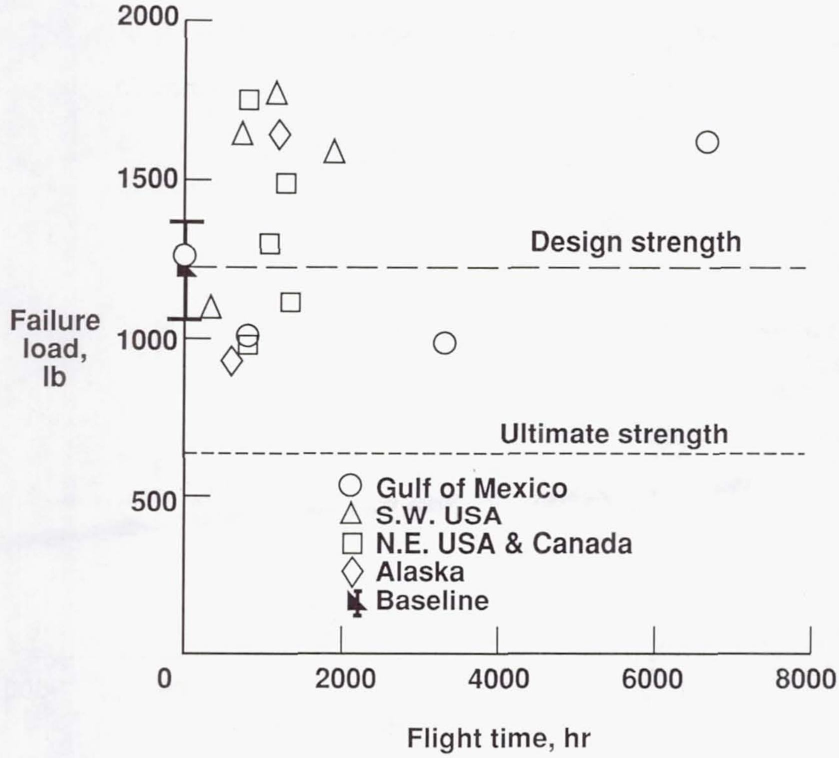


Figure 9. Failure load of litter doors after exposure.

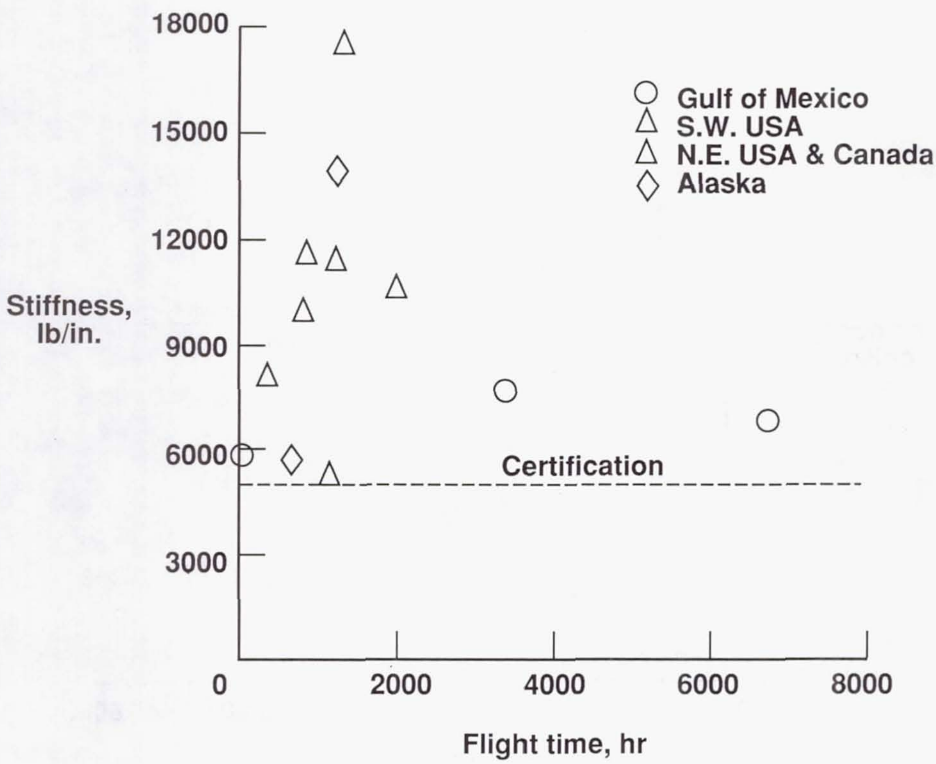


Figure 10. Stiffness of litter doors after exposure.

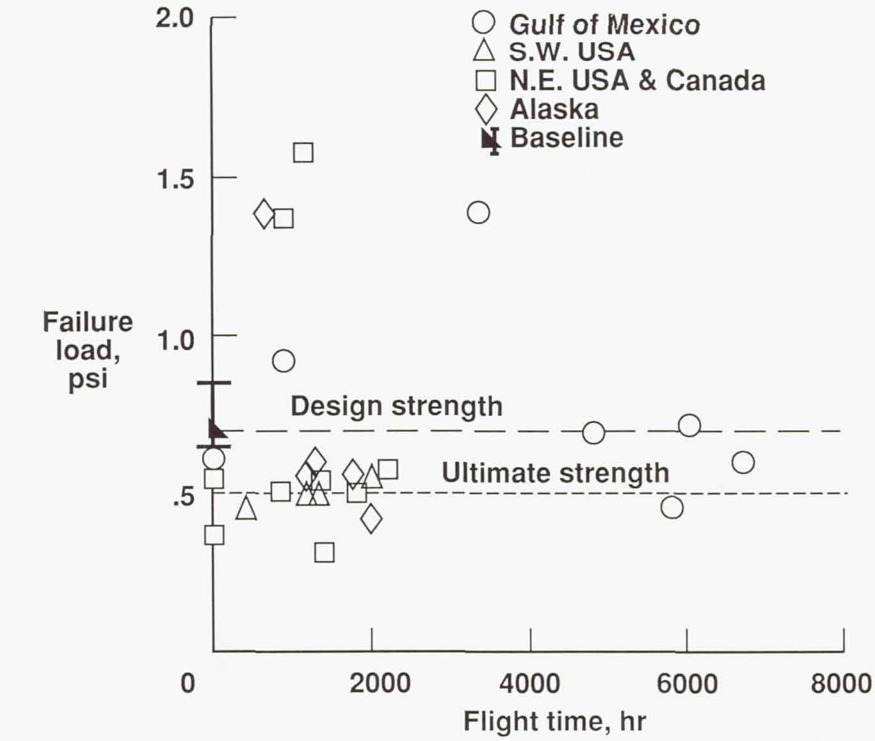


Figure 11. Failure load of baggage doors after exposure.

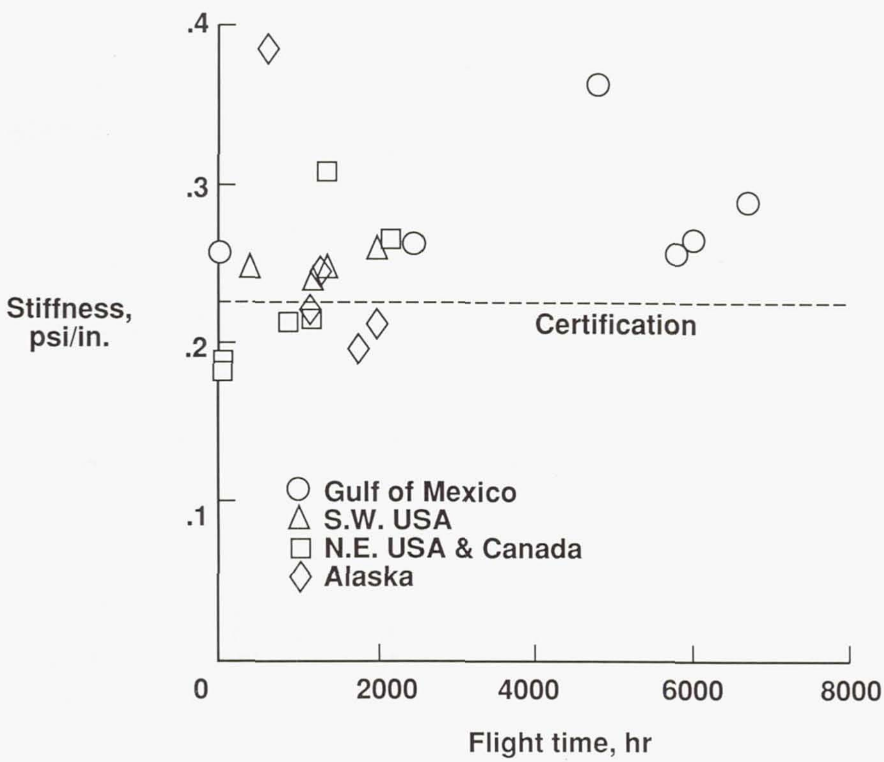


Figure 12. Stiffness of baggage doors after exposure.

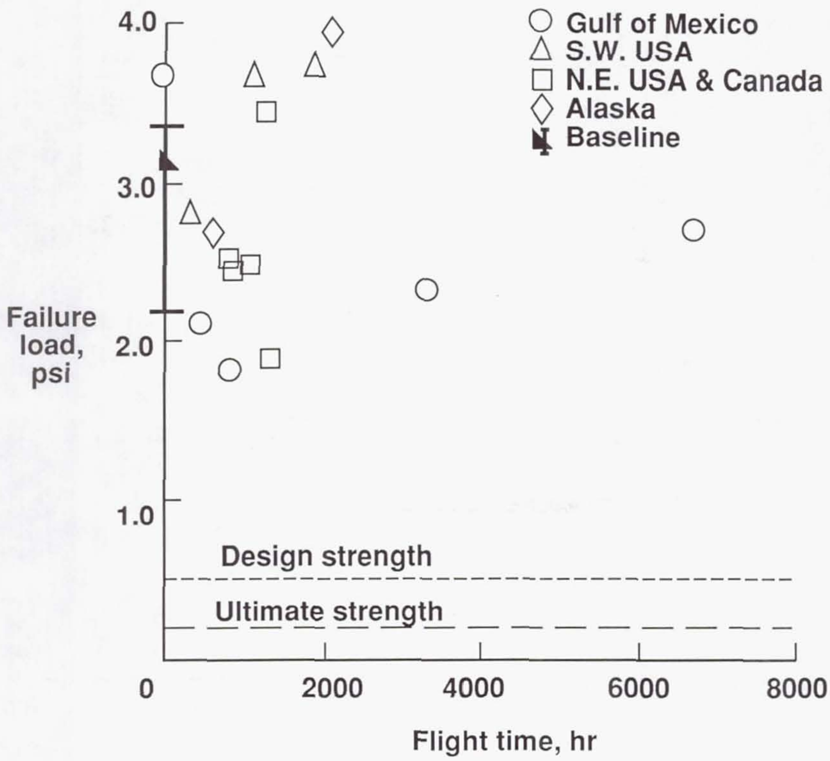


Figure 13. Failure load of forward fairings after exposure.

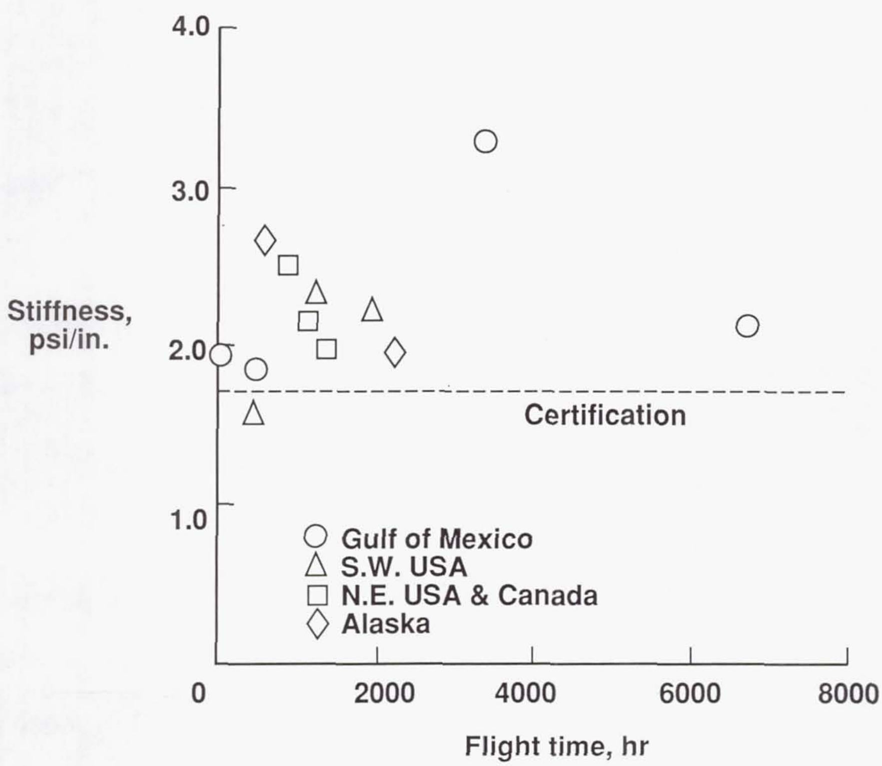


Figure 14. Stiffness of forward fairings after exposure.

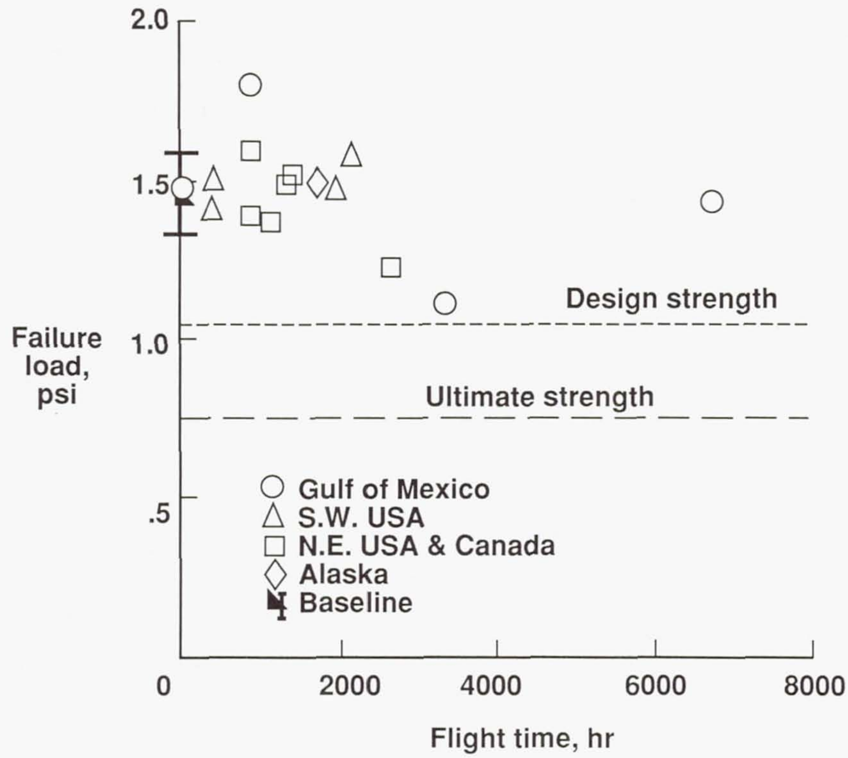


Figure 15. Failure load of vertical fins after exposure.

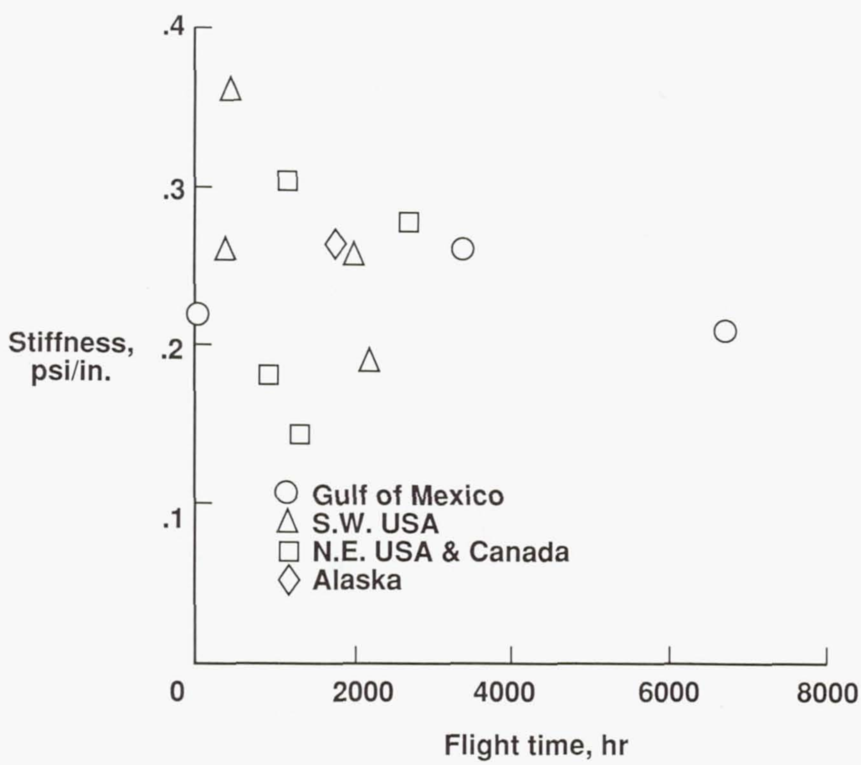


Figure 16. Stiffness of vertical fins after exposure.

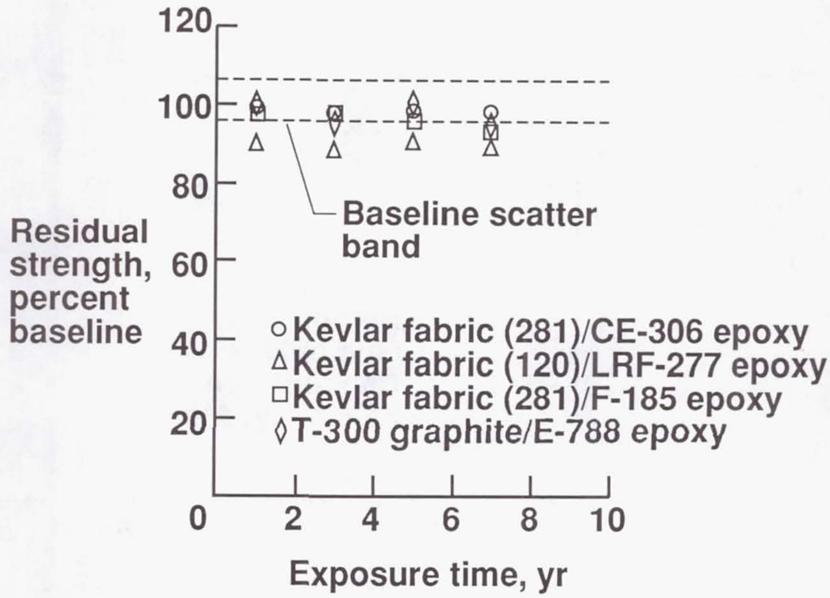


Figure 17. Residual compressive strength of composite materials after exposure.

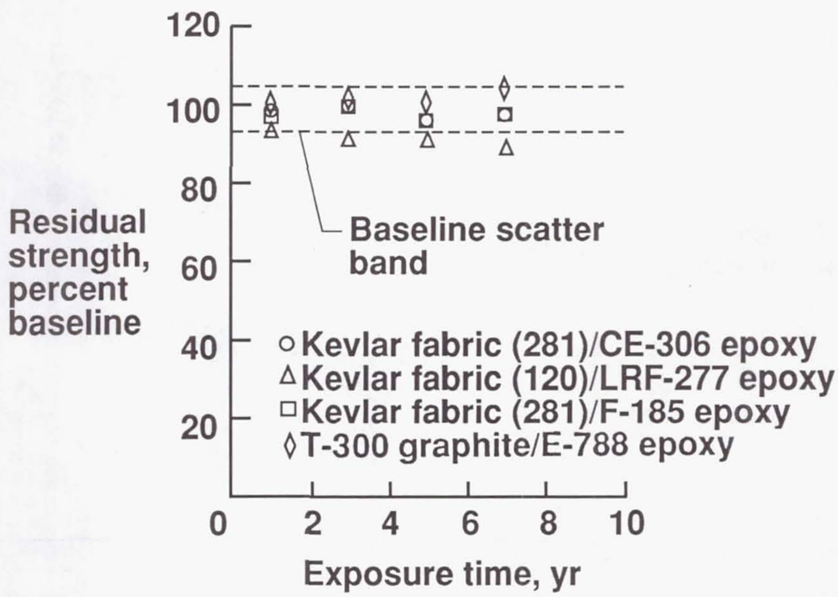


Figure 18. Residual short-beam shear strength of composite materials after exposure.

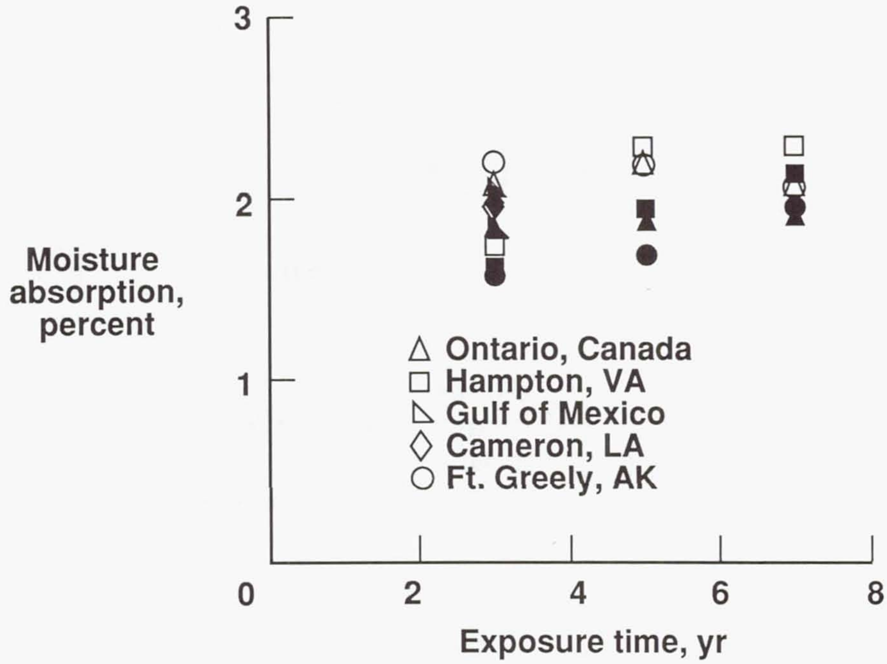


Figure 19. Moisture absorption of Kevlar-49/CE-306 composite material. Solid symbols represent unpainted specimens.

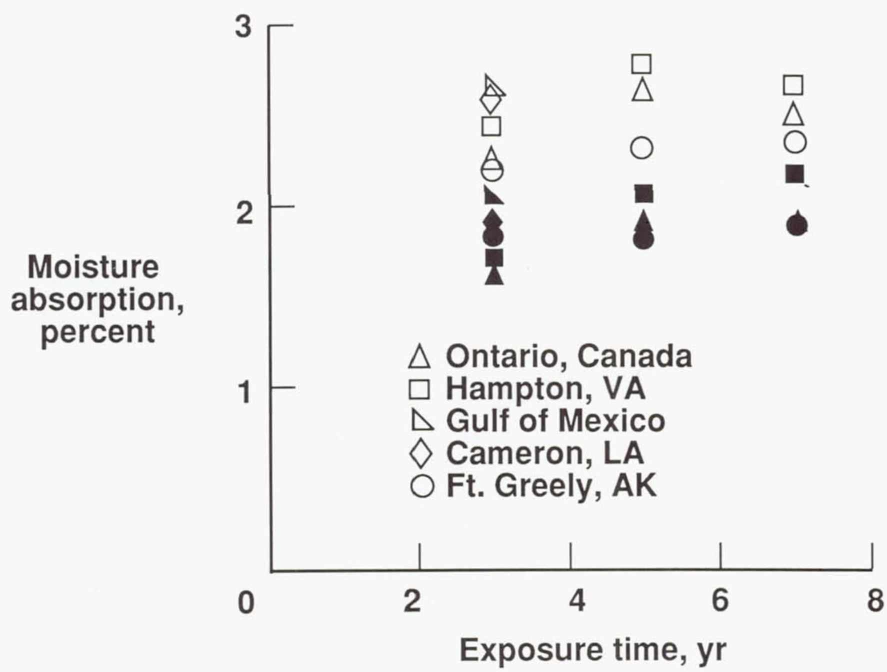


Figure 20. Moisture absorption of Kevlar-49/F-185 composite material. Solid symbols represent unpainted specimens.

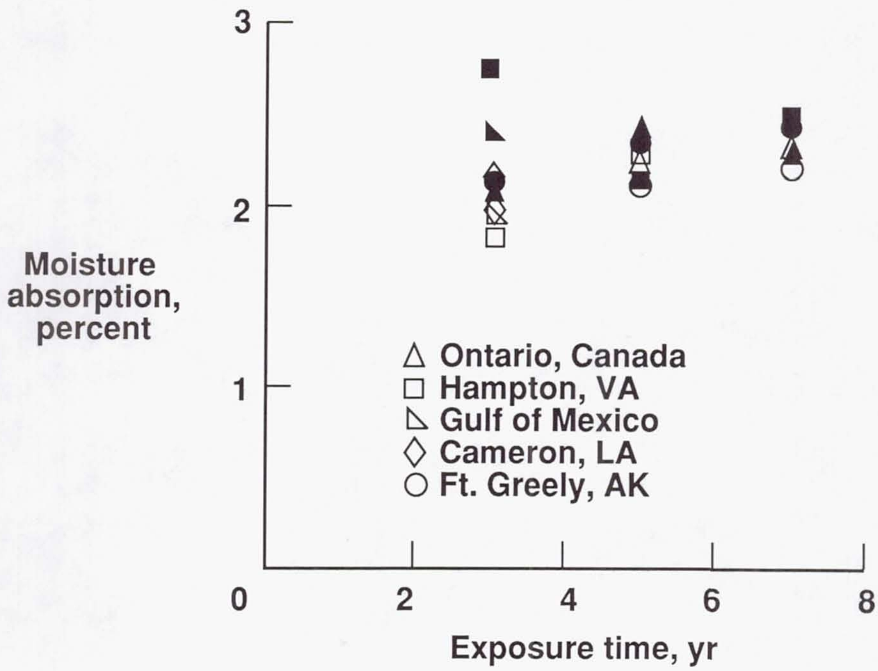


Figure 21. Moisture absorption of Kevlar-49/LRF-277 composite material. Solid symbols represent unpainted specimens.

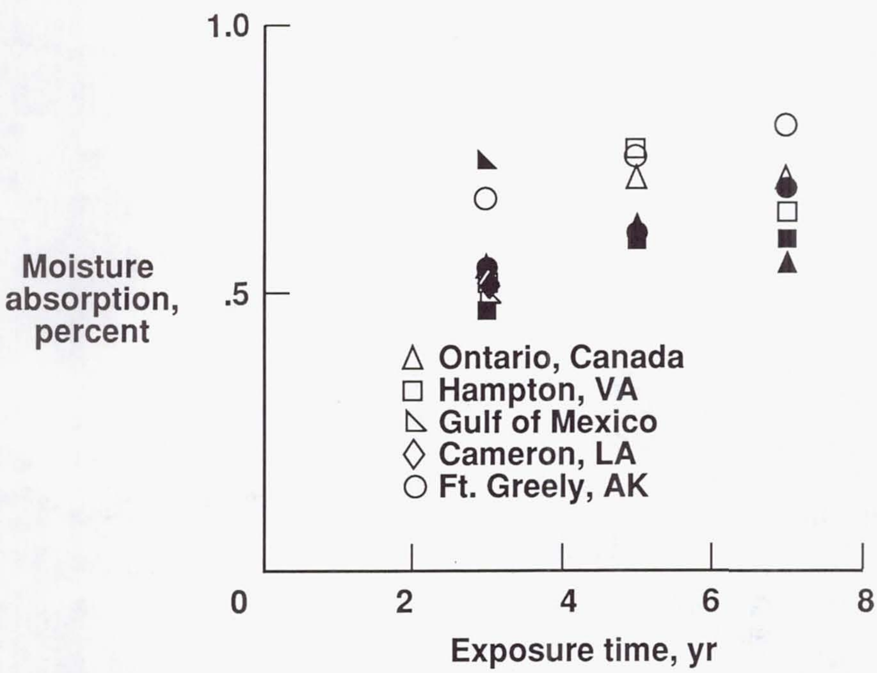
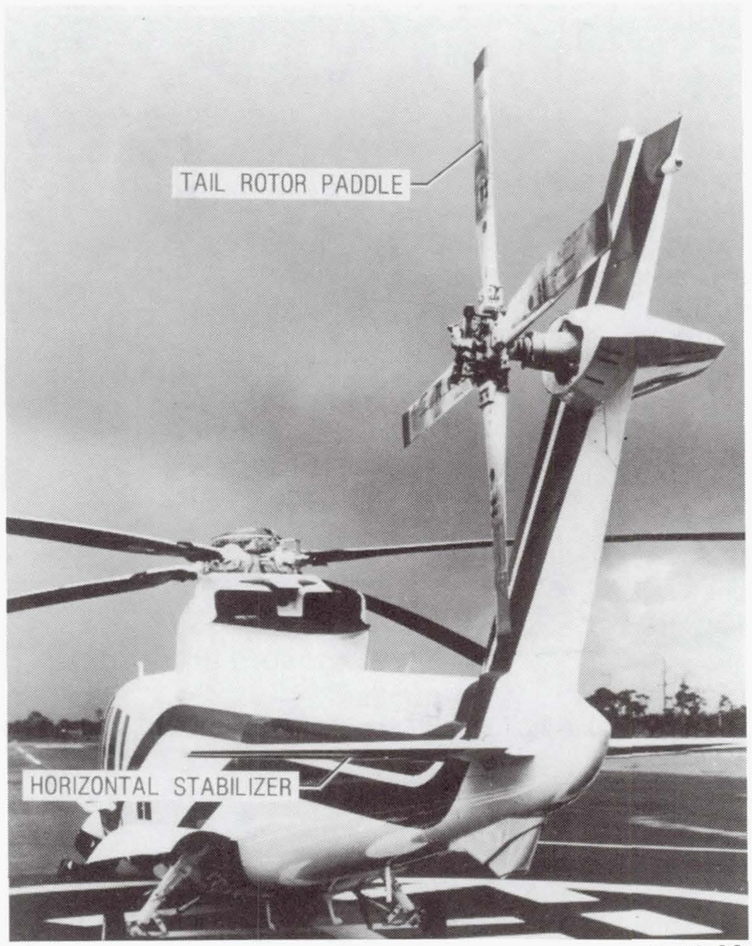


Figure 22. Moisture absorption of T-300/E-788 graphite/epoxy composite material. Solid symbols represent unpainted specimens.



L-90-36

Figure 23. Composite components in flight service on Sikorsky S-76 helicopter.

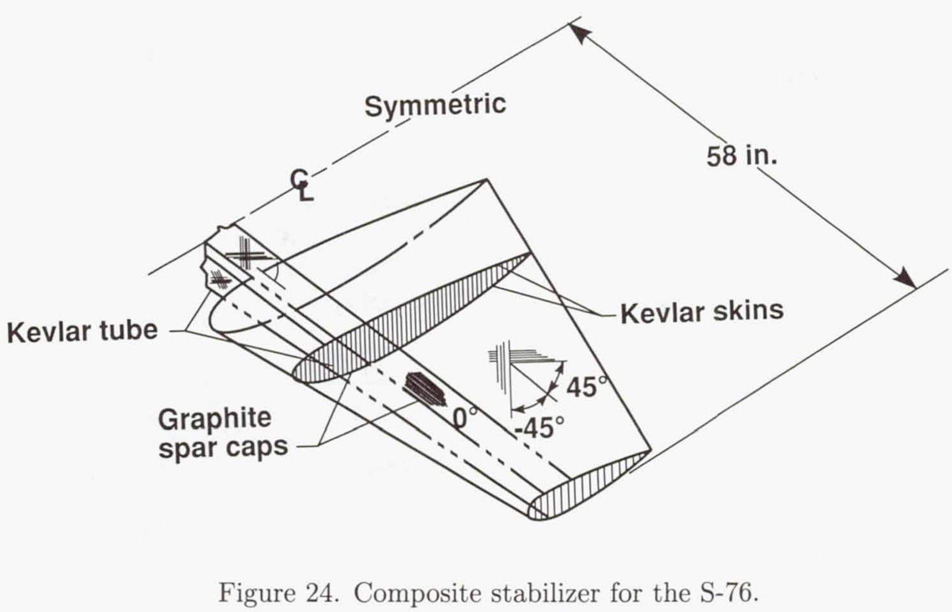
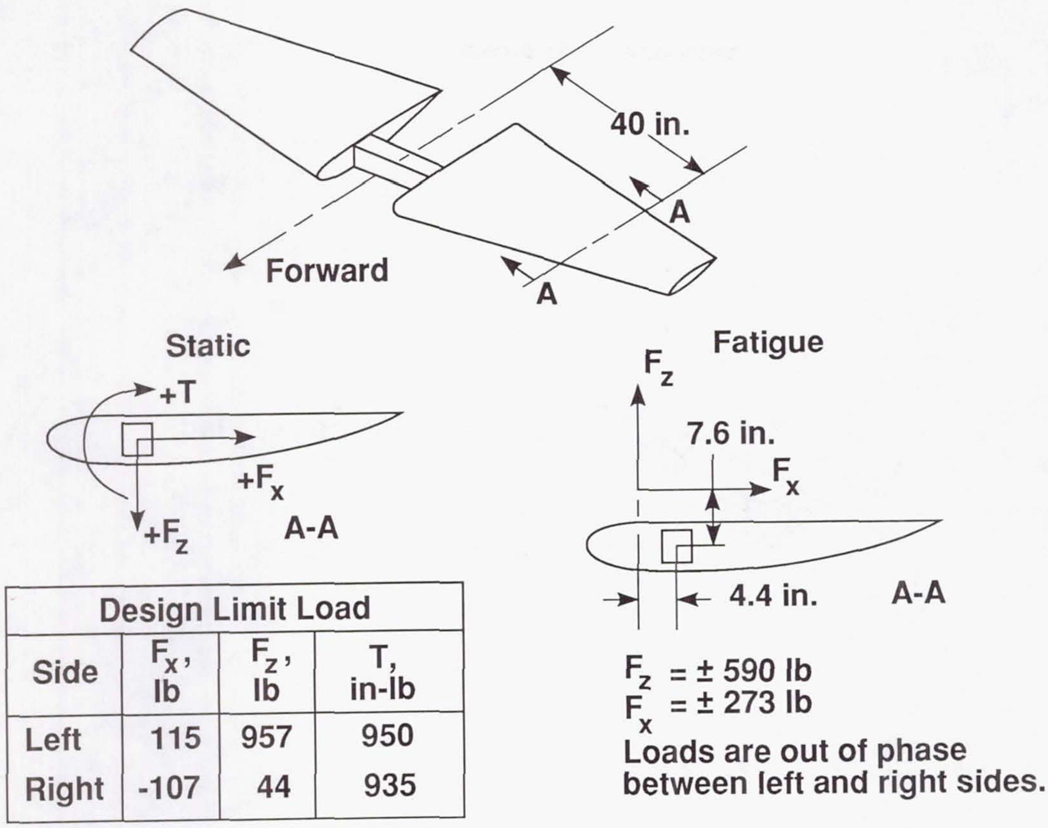


Figure 24. Composite stabilizer for the S-76.



Design Limit Load			
Side	F_x, lb	F_z, lb	T, in-lb
Left	115	957	950
Right	-107	44	935

Figure 25. Load conditions for the S-76 horizontal stabilizer.

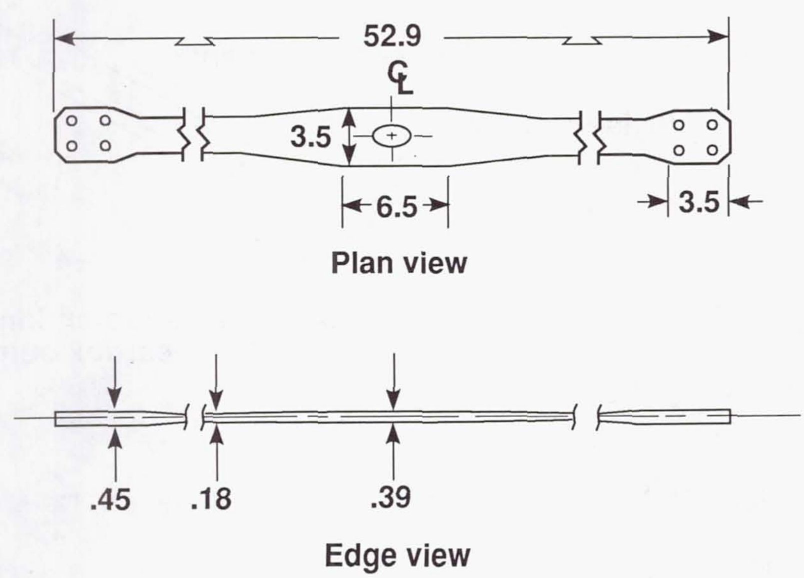


Figure 26. Composite tail rotor spar for the S-76. All dimensions are given in inches.

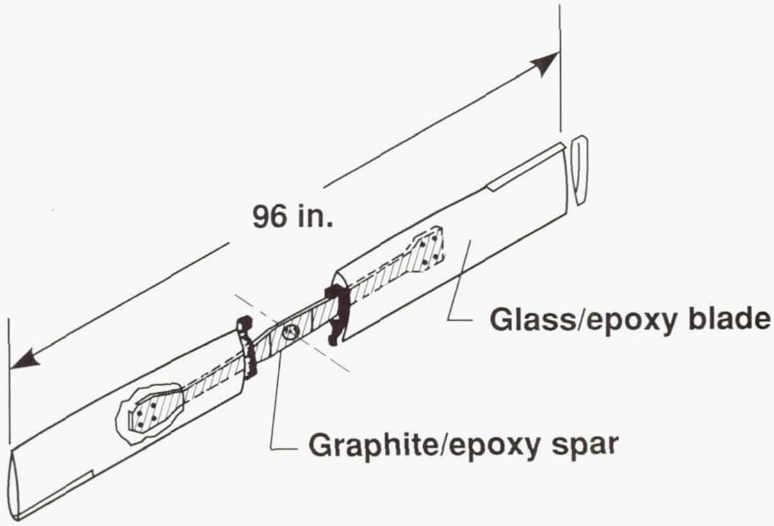


Figure 27. Sikorsky S-76 tail rotor paddle.

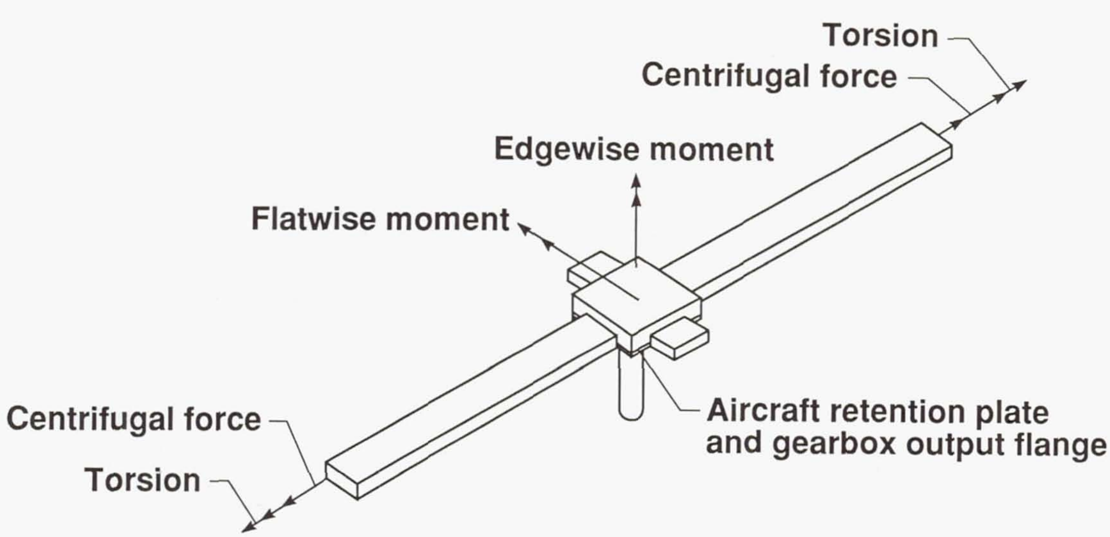


Figure 28. Schematic diagram of S-76 tail rotor spar loadings.

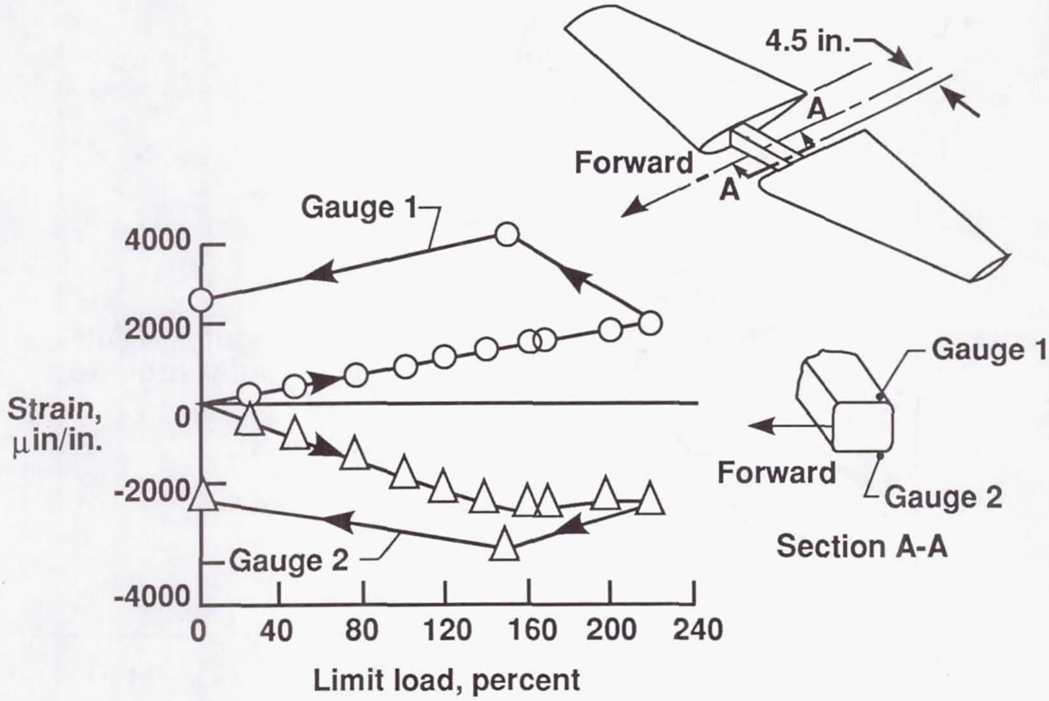


Figure 29. Strain as a function of limit load on the S-76 stabilizer.

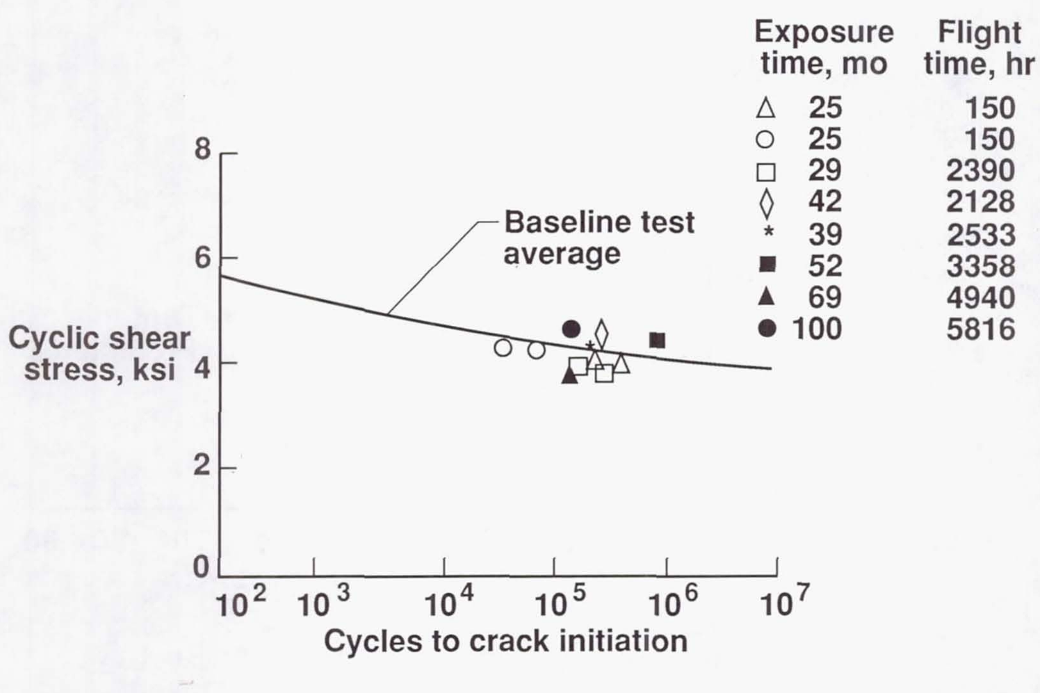


Figure 30. Effect of service environment on S-76 composite tail rotor spars.

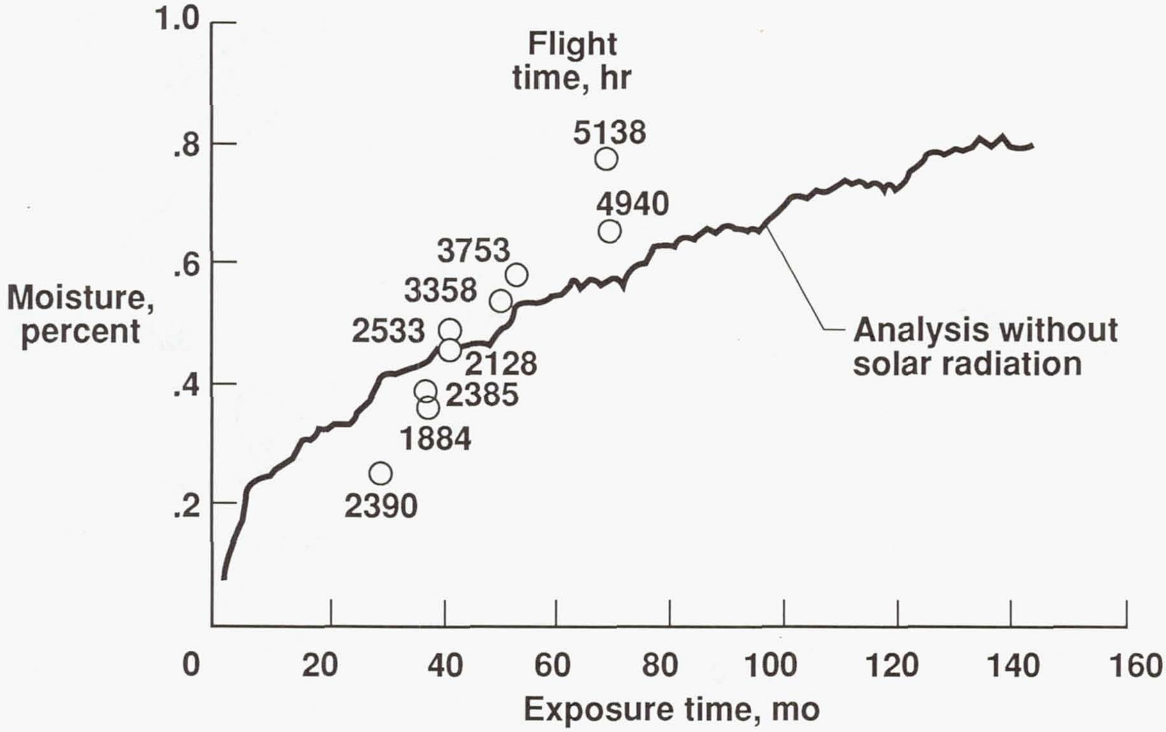


Figure 31. Measured and predicted moisture content for S-76 tail rotor spars exposed at Lake Charles, Louisiana.

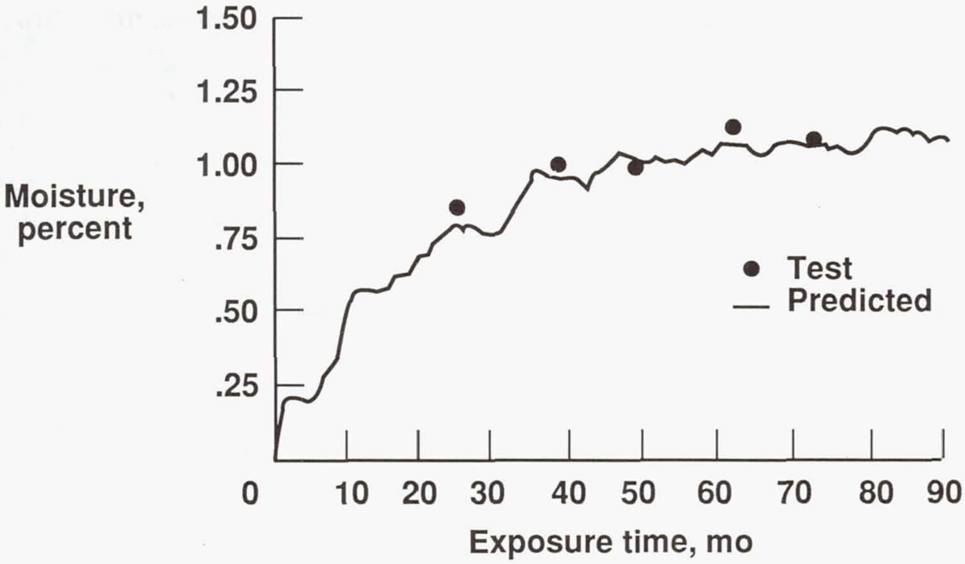


Figure 32. Measured and predicted moisture content for 6-ply AS1/6350 material exposed at Stratford, Connecticut.

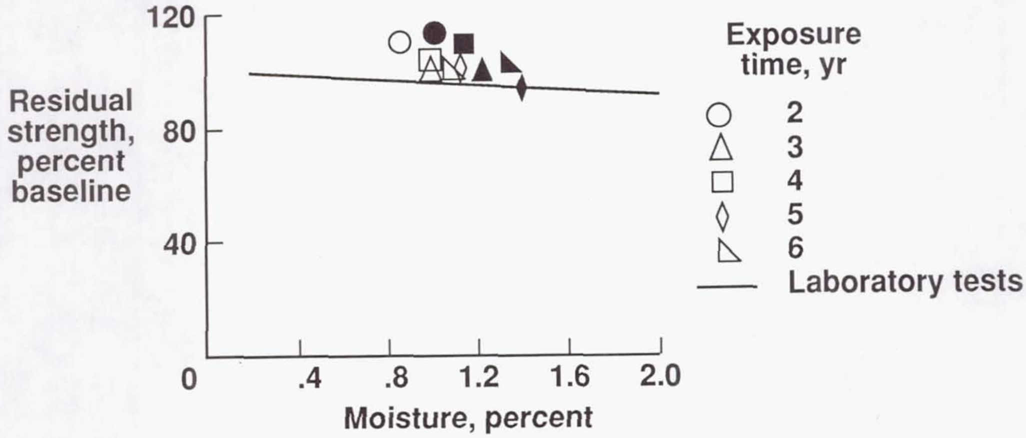


Figure 33. Effect of moisture on residual flexure strength of 6-ply AS1/6350 material. Open symbols represent Stratford, Connecticut; solid symbols represent West Palm Beach, Florida.

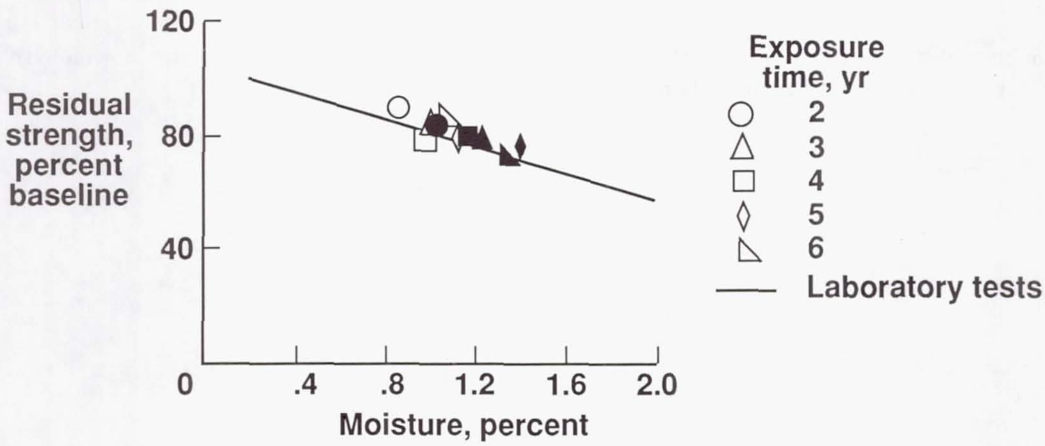


Figure 34. Effect of moisture on residual short-beam shear strength of 6-ply AS1/6350 material. Open symbols represent Stratford, Connecticut; solid symbols represent West Palm Beach, Florida.

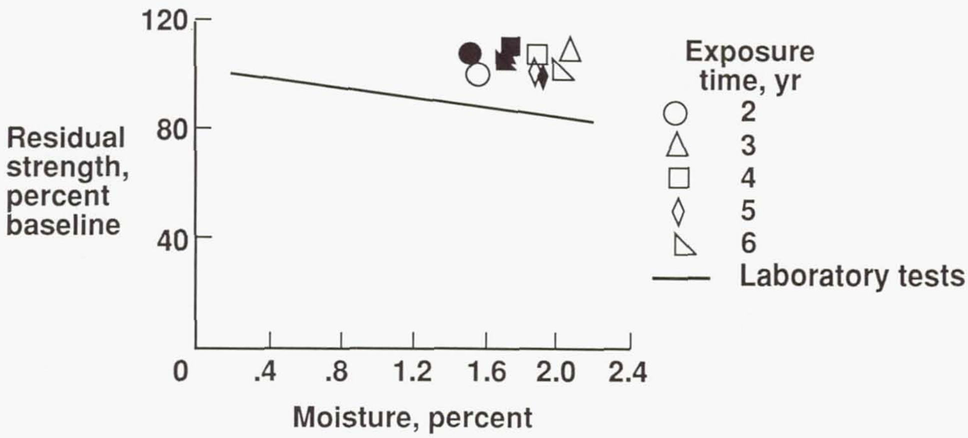


Figure 35. Effect of moisture on residual tension strength of 6-ply AS1/6350 material. Open symbols represent Stratford, Connecticut; solid symbols represent West Palm Beach, Florida.

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Report Documentation Page

1. Report No. NASA TM-4195 AVSCOM TR-90-B-004	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle Evaluation of Composite Components on the Bell 206L and Sikorsky S-76 Helicopters		5. Report Date August 1990
		6. Performing Organization Code
7. Author(s) Donald J. Baker		8. Performing Organization Report No. L-16776
9. Performing Organization Name and Address Aerostructures Directorate USAARTA-AVSCOM Langley Research Center Hampton, VA 23665-5225		10. Work Unit No.
		11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546-0001 and U.S. Army Aviation Systems Command St. Louis, MO 63120-1789		13. Type of Report and Period Covered Technical Memorandum
		14. Sponsoring Agency Code 1L162211A47AB

15. Supplementary Notes

Donald J. Baker: Aerostructures Directorate, USAARTA-AVSCOM.

Presented at the Eighth DoD/NASA/FAA Conference on Fibrous Composites in Structural Design, Norfolk, Virginia, Nov. 28–30, 1989.

16. Abstract

Progress is described on two programs to evaluate structural composite components in flight service on Bell 206L and Sikorsky S-76 commercial helicopters. Forty ship sets of composite components that include the litter door, baggage door, forward fairing, and vertical fin have been installed on Bell 206L helicopters that are operating in widely different climates. Component installation started in 1981, and selected components are being removed and tested at prescribed intervals over a 10-year evaluation. Four horizontal stabilizers and 11 tail rotor spars that are production components on the S-76 helicopter are being tested after prescribed periods of service to determine the effects of the operating environment on their performance. Concurrent with the flight evaluation, the materials used to fabricate the components are being exposed in ground racks and tested at specified intervals to determine the effects of outdoor environments. In this paper, results achieved from 123 000 flight hours of accumulated service on the Bell 206L components and 53 000 flight hours on the Sikorsky S-76 components are reported. Seventy-eight Bell 206L components have been removed and tested statically. Results are presented after 7 years of ground exposure of materials used to fabricate the Bell 206L components. Results of tests on 4 Sikorsky S-76 horizontal stabilizers and 11 tail rotor spars are presented. Panels of material used to fabricate the Sikorsky S-76 components that were exposed for 6 years have been tested and results are presented.

17. Key Words (Suggested by Authors(s)) Composite materials Environmental effects Flight service	18. Distribution Statement Unclassified—Unlimited Subject Category 24

19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 35	22. Price A03

NASA FORM 1626 OCT 86 NASA-Langley, 1990

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