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Flight Service Evaluation of Composite Components on the Bell Helicopter Model 206L: Design, Fabrication, and Testing

Herbert Zinberg

Bell Helicopter Textron Inc.
Fort Worth, Texas

Contract NAS1-15279
November 1982

NASA
National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

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FOREWORD*

This report was prepared by Bell Helicopter Textron Inc., under contract NAS1-15279, "Flight Service Evaluation of Composite Components on the Bell Helicopter Model 206L," and covers the work performed from November 1978 through June 1982. This work encompasses the design, fabrication and test phases of the program.

The program was jointly funded by the Materials Division of the NASA-Langley Research Center and the Structures Laboratory, USARTL (AVRADCOM). The NASA-Langley Technical monitor for this contract was Mr. Donald J. Baker. The Bell Helicopter Textron Project Engineer was Mr. Herbert Zinberg.

Acknowledgement is made of the work performed by the following Bell Helicopter personnel: L. Williams, Research Projects; D. Crist, Airframe Design; J. Harse, Research Projects.

Certain materials are identified in this publication to specify adequately which materials were investigated. In no case does such identification imply recommendation or endorsement of the material by NASA or USARTL (AVRADCOM), nor does it imply that the materials are necessarily the only ones or the best ones available for the purpose.

*The contract research effort which has led to the results in this report was financially supported by the Structures Laboratory, USARTL (AVRADCOM).

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SUMMARY

This report describes the design, fabrication, and testing phases of a program to obtain long term flight service experience on four representative helicopter airframe structural components operating in typical commercial environments. The aircraft chosen for this program is the Bell Helicopter Model 206L. The structural components are the forward fairing, litter door, baggage door, and vertical fin. The advanced composite components were designed to replace the production parts in the field and were certified by the FAA to be operable through the full flight envelope of the 206L.

Forty sets of flight service components and ten sets of certification test parts were fabricated in the Bell production facilities. The forty sets will be located throughout the forty eight contiguous states, Alaska, and Canada. In addition, a total of approximately two thousand tensile, compression, and short beam shear specimens were built and installed on exposure racks placed in locations having the same types of environment as the test helicopters.

A description of the fabrication process that was used for each of the components is given in this report. As part of the fabrication program, a cost tracking system was instituted in order to develop data to predict production costs. Within the number of parts built, no significant cost reduction trends were found. It is probable that the samples were not sufficiently large to develop cost trends.

In order to receive FAA certification, it was necessary to perform static failing load tests on all four components. In addition fatigue tests were run on four specimens that simulated the attachment of the vertical fin to the helicopter's tail boom.

After receiving FAA certification, the forty sets of components were delivered to eleven commercial operators for service evaluation. At the end of the first, third, and fifth years of flight a scheduled number of components will be returned to Bell for static test. Concurrently, a specified number of the exposure specimens will be sent to NASA, Langley Research Center for test. The results of these tests will provide data on the serviceability of advanced composite airframe structures in the commercial helicopter environment.

1. INTRODUCTION

The introduction of advanced composite materials into fixed wing aircraft has proceeded in an orderly manner during the past few years. Military and large commercial airplanes are now taking advantage of the unique characteristics of these materials to reduce both weight and cost. However, before advanced composite materials became accepted by the airplane operators, it was necessary to perform a number of programs to evaluate these materials in actual service. These programs, sponsored by NASA and the military, revealed no major structural or maintenance problem on either military or commercial airplanes.

The advantages of using advanced composite materials on light commercial helicopters are obvious, but until serviceability is demonstrated, many helicopter operators are reluctant to replace well-proven metals with composites. In many respects the operational environments for light commercial helicopters can be more hostile than for fixed wing aircraft. Helicopters often operate for long periods of time in remote areas where maintenance facilities are, at best, primitive; and because helicopters often operate near unimproved ground, damage from tree limbs, rocks, sand, and other debris is common. Accordingly, a program to demonstrate the serviceability of advanced composite airframe structures in the environment of light commercial helicopters is required before these materials will be fully accepted for production. This report describes a portion of such a program being performed by Bell Helicopter Textron Inc. (BHTI) under the joint sponsorship of NASA Langley Research Center and the Structures Laboratory, U.S. Army Research and Technology Laboratories (AVRADCOM).

The helicopter chosen for this program is the BHTI Model 206L LongRanger. This helicopter, and its predecessor the JetRanger, has been in service throughout the world for over 10 years, and has a well-established service record. A direct comparison can therefore be made between composite parts that are substituted for production metal parts on a one-to-one basis. Another reason for choosing the Model 206L is that these aircraft are operating in sufficiently diverse areas of the United States and Canada in large enough numbers that a representative sampling of helicopters (and cooperating operators) could be found in each of the operating environments in which service evaluation is desired.

The overall program consists of two separate phases. The first phase involves design, fabrication, and structural testing of the composite components. The second phase involves flight service evaluation and post service testing of the components.

An overview of the total program is given in Section 2. The remainder of this report describes the activities of the first phase, the work leading up to the installation, and flight service evaluation of the composite components.

2. PROGRAM OVERVIEW

The objective of this program is to evaluate the effects of long-term service experience on helicopter airframe components made from advanced composite materials. It is anticipated that knowledge will be gained concerning the degredation of strength over long periods of time. Also, a qualitative assessment will be made of the maintainability and overall serviceability of the components. To accomplish the stated objectives, the following requirements are set forth for the program:

a. The components chosen for evaluation must form a part of the external surface of the aircraft so that they are continuously exposed to the aircraft's external environment. Also, the components must be such that a structural failure of any of them could not result in loss of the helicopter.

b. The components have to be certified by the FAA for unrestricted operation within the full operational envelope of the helicopter.

c. The components have to be built in a production environment, using production tools, and receive no special considerations other than those normally required to produce flight quality hardware.

d. The components have to be installed on a large enough sampling of helicopters, operating in sufficiently diverse environments, that a complete range of operating environments are encountered over a long period of time.

The four components chosen for evaluation in this program are the forward fairing, litter door, baggage door, and vertical fin. These are shown in their respective locations in Figure 1. The forward fairing, litter door, and vertical fin were designed and built by BHTI. The baggage door was designed and built by the Brunswick Defense Division of Lincoln, Nebraska. The two doors and fairing are made primarily from Kevlar-49/epoxy fabric. The vertical fin is made primarily from graphite/epoxy prepreg tape.

The FAA certification program required that each component be static tested to failure, and that the vertical fin be both static and fatigue tested. The Kevlar components were tested at room temperature conditions. Factors were developed by which the room temperature strength had to exceed design loads to account for the reduction in material strength caused by temperature and

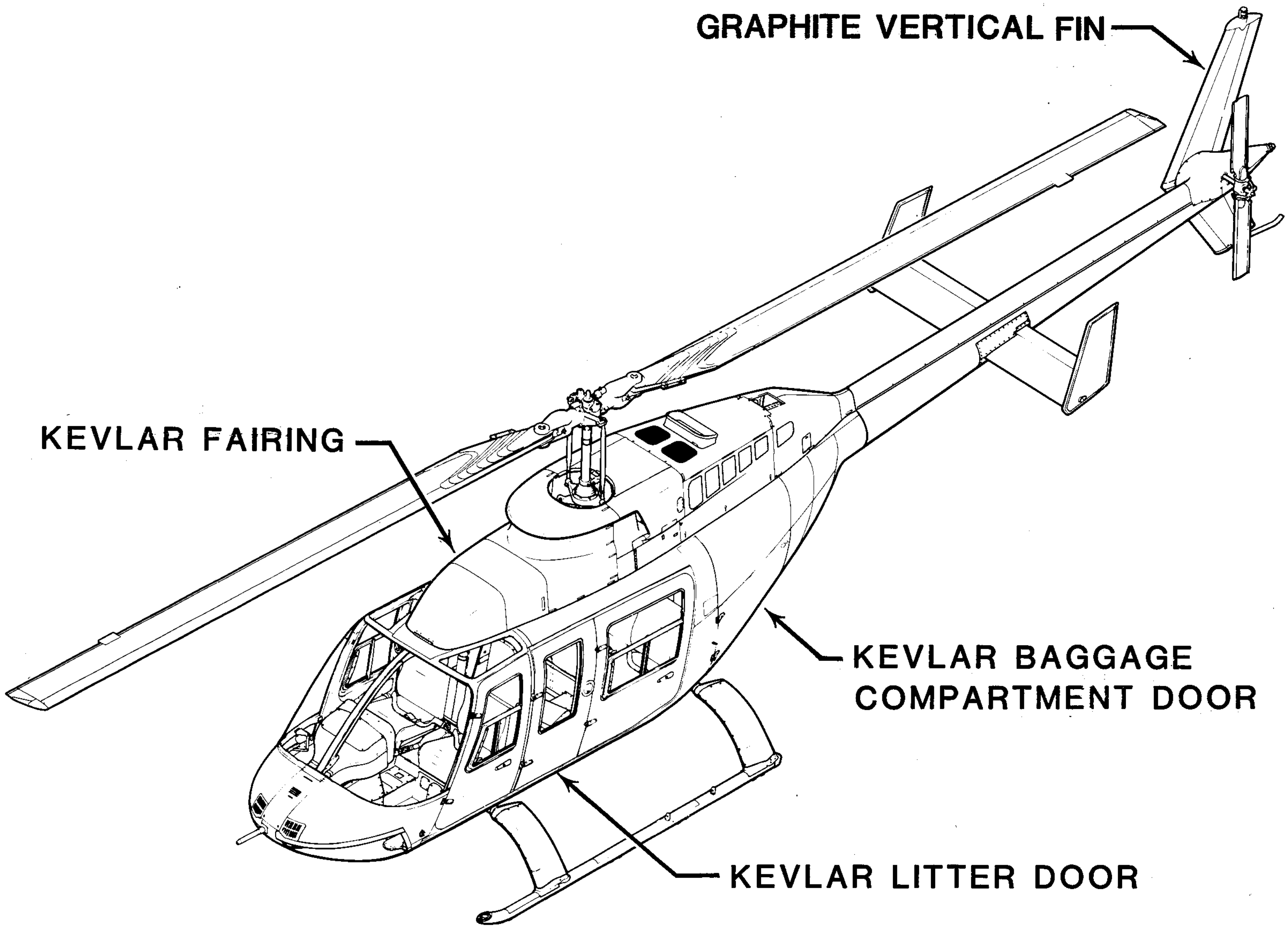

Figure 1. Bell LongRanger helicopter service evaluation components.

moisture. This was done by testing a series of coupons at environmentally soaked conditions and comparing them with room temperature strengths. The vertical fin was environmentally conditioned just prior to testing.

The design, tooling, and certification of the two doors and vertical fin were done under BHTI and Brunswick Independent Research and Development programs. This included the fabrication of a total of ten components for FAA certification. The tools were built as semiproduction tools, capable of fabricating approximately 100 units. Since the components and tools received FAA conformity inspection, no major design, such as changes in fabrication procedures, were permited since these would require reapproval from the FAA. Also, since the conponents were made in the production shop, it was possible to institute a cost tracking system. The costs are discussed in Section 5.6.

As part of the fabrication program, five of each production component were picked at random and static tested to determine the scatter in strength, and to compare the strength with the certification components. The results of these tests are given in Section 6.2.5.

The components are scheduled to be installed on a total of 40 helicopters located in the Gulf Coast area, Northeast United States, Southeast Canada, and Alaska. Table 1 shows the specific locations of the 40 sets of components. Most of them will be flown for 5 years with an option of an additional 5 years. Some components will be removed at prescribed time periods (one, three and five years) and returned to BHTI for ultimate static load tests. These test results will be compared with the tests conducted prior to flight sevice to determine the effect that prolonged commercial service has on the strength of the materials.

To supplement the flight data, a total of approximately 2000 tension, compression, and short beam shear test coupons representative of the components' laminates have been installed on five exposure racks. The racks are located on an oil platform in the Gulf of Mexico, at Cameron, Louisiana, NASA Langley, Hampton, Virginia, Toronto, Ontario, and Fort Greeley, Alaska. One-fifth of the specimens on each rack will be removed after one (1), three (3) and five (5) years of exposure and sent to NASA Langley Research Center for test.

It is believed that this extensive service experience, supplemented by the component and coupon tests, will provide essential data on the reliability, maintainability, and overall structural behavior of advanced composite materials in actual commercial service.

TABLE 1. DISTRIBUTION OF FLIGHT SERVICE EVALUATION SETS

Operator	Location	No. of Sets
Gulf Coast		
Petroleum Helicopters	Lafayette, La.	5
Commercial Helicopters	New Iberia, La.	2
Air Logistics	New Iberia, La.	5
Houston Helicopters	Houston, Tx.	3
Northeast U.S./Southeast Canada		
Royal Canadian Mounted Police	Ottawa, Ont.	2
Canadian Ministry of Transportation	Ottawa, Ont.	2
Trans-Quebec Helicopters	Montreal, Que.	5
Heli-Voyageur Helicopters	Val d 'Or, Que.	3
Island Helicopters	Garden City, N. Y.	3
Alaska		
ERA Helicopters	Anchorage, Alaska	5
Southwest U.S.		
Air Services International	Scottsboro, Ariz.	5

3. DESCRIPTION OF COMPONENTS

The four components being evaluated in this program - forward fairing, litter door, baggage door, and the vertical fin - are described in this section.

3.1 FORWARD FAIRING

Figures 2a and 2b show the outer and inner views, respectively, of the fairing. The fairing is 35.9 in. long, and is 29 in. wide and 13 in. high at the aft end. It is attached to the roof structure by a hinge at the forward end and two latches at the aft end. The latches are shown in Figure 2b, but the hinge, which is installed in the field, is not. Both the hinge and latches are the same as those used on the metal production fairings. A rubber seal, located between the fairing and the roof, is compressed when the fairing is closed. The stays shown in Figure 2b are used to hold the fairing open while the aircraft is being serviced.

The fairing has a single curvature at its aft end that changes to a severe double curvature in the vicinity of the forward end. It is a sandwich structure that uses a single ply of 281 style Kevlar/epoxy fabric with CE306 epoxy[1] for both the inner and outer facesheets. The core is 0.38-in.-thick, 4.5-lb/ft^3 density foam. The foam is Klegecell[2] - a closed-cell, thermoplastic polyvinylchloride foam that can be preformed under heat and pressure. It has moderate strength up to 200°F and is moisture-resistant. The principal reason for using it, however, is its low cost.

A feature of the forward fairing is that no adhesive, other than the epoxy matrix, is required to bond the foam core to the facesheets. A single ply of 281 style fabric with the 0.38-in. thick core is sufficient to achieve the required strength and stiffness.

Details of the fabrication process for making the forward fairing are given in Section 5.1.

3.2 LITTER DOOR

The litter door, shown in Figures 3a and 3b, is 26 in. wide by 46 in. high, and is located on the left-hand side of the aircraft

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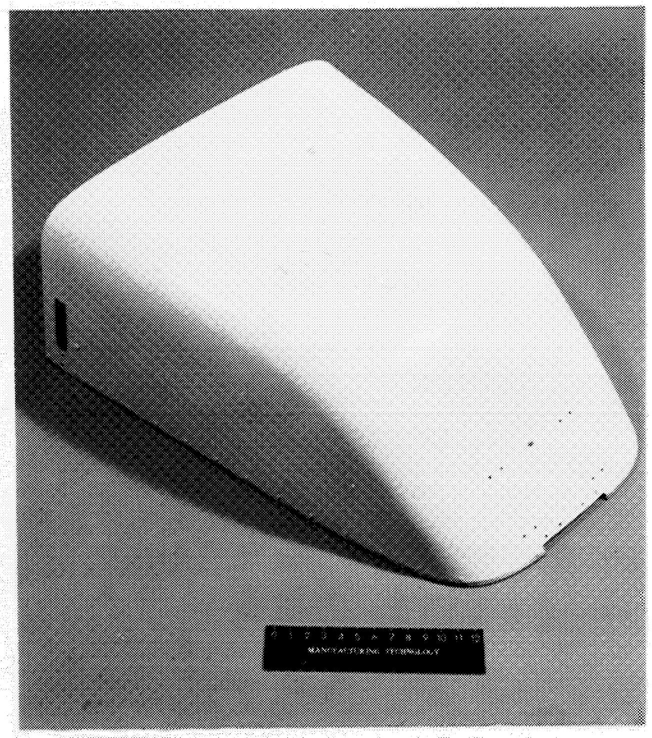
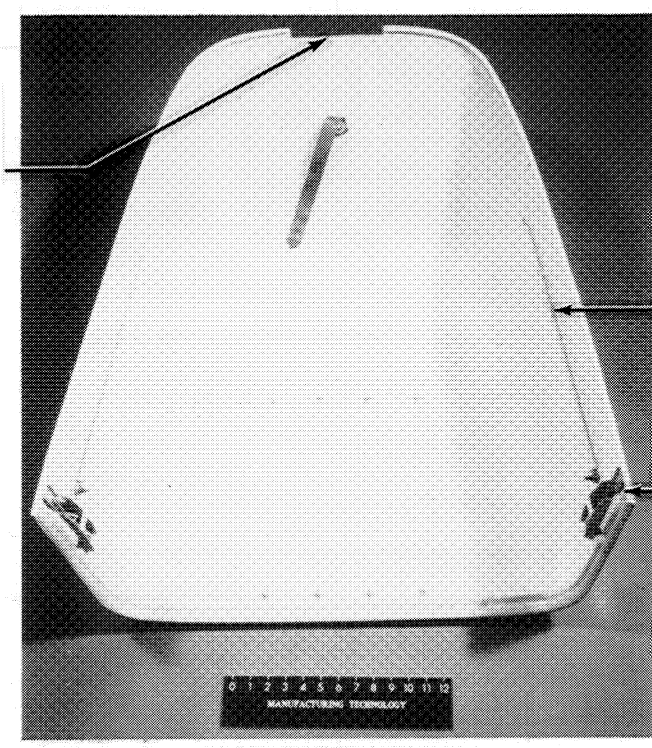
[1]Manufactured by Ferro Corporation, Culver City, CA.
[2]Manufactured by the Klegecell Corporation, Grapevine, TX.

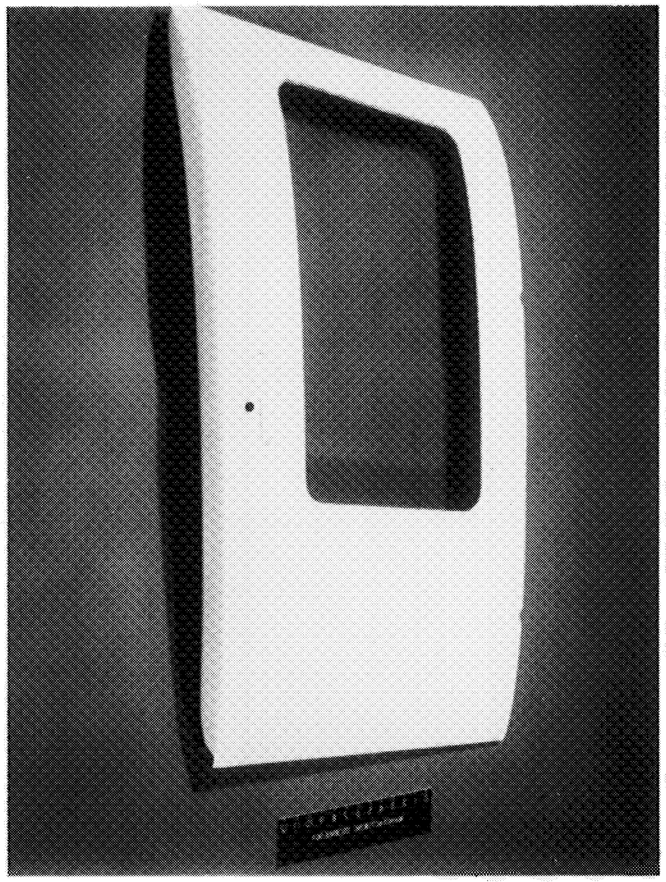


a. Outer view

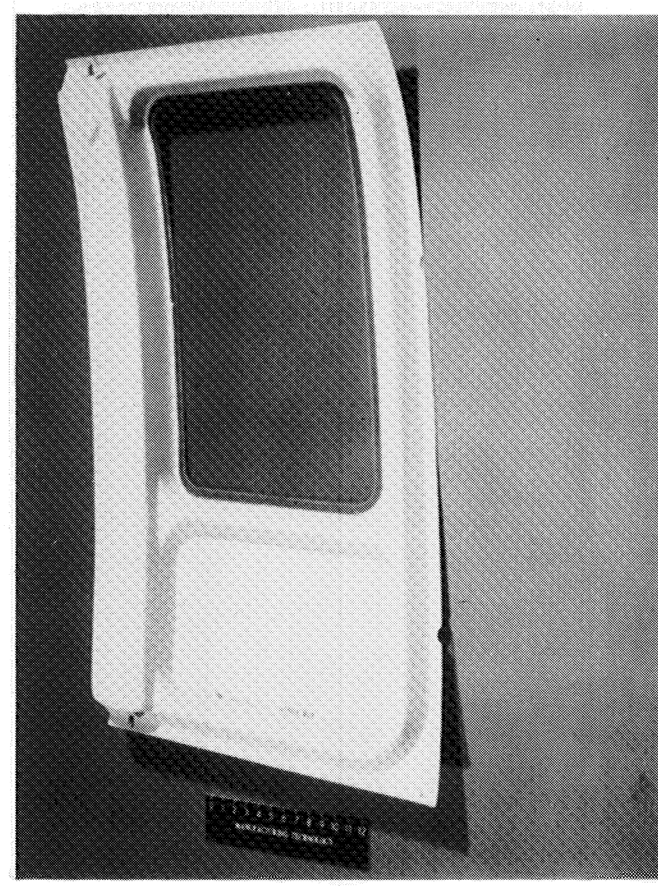


b. Inner view

Figure 2. Kevlar/epoxy forward fairing.



a. Outer view



b. Inner view

Figure 3. Composite litter door.

between the crew and cabin doors. The cabin door is hinge-mounted on the aft edge of the litter door, which in turn is mounted on the airframe by two hinges at its forward edge. In normal operation, the litter door acts as a fixed panel. When a litter or oversized cargo is loaded, the cabin door is opened, the litter door latch is opened, and both doors are swung forward, with the litter door's forward hinges supporting both doors. Figure 4 shows the litter and cabin doors in the open position.

Figure 4. Litter and cabin doors in open position.

Kevlar/epoxy fabric and tape are the composite materials used for the door. F560 resin[3] was used with the tape, and F185[3] with the fabric. As shown in Figure 3b, the structure is composed of an outer skin, an inner skin in the form of a continuous hat section, and a door post. The outer skin is made from two plies of 281 style fabric and one ply of 220 style fabric. The inner skin is three plies of 281 style fabric reinforced with unidirectional Kevlar/ epoxy tape.

The plexiglass window is bonded directly to the door; no edging is required. The latches are located inside the door post and are actuated by a flush handle. The latching mechanism is accessible from the outside by removing the outer latch plate.

[3]Manufactured by the Hexcel Corporation, Dublin, CA.

The aft hinge half is shown in Figure 5. It is installed from the exterior of the door by screwing it into the threaded pin shown in the figure. The same part is used on both the upper and lower hinges. Two ramps are machined into it to form the stop surfaces for the cabin door.

Custom panels are provided to cover the inner surface of the door when the aircraft is used as an executive transport. These panels may be installed at the option of the operator.

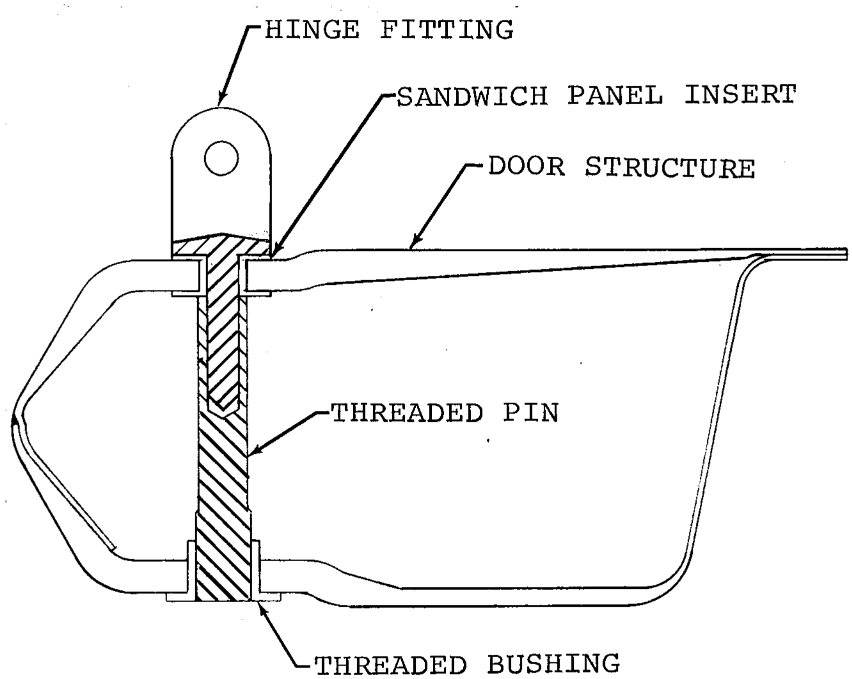


Figure 5. Litter door lift hinge installation.

3.3 BAGGAGE DOOR

The baggage door is located on the left-hand aft section of the fuselage, as shown in Figure 1. The door is a sandwich structure made from Kevlar/epoxy fabric[4] with a honeycomb core. It is attached to the fuselage by two metal hinges at the forward end and two quick-release latches at the aft end. A key-operated lock is also located at the aft end of the door.

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[4]The matrix for the Kevlar/epoxy is a proprietary resin system manufactured by the Brunswick Defense Division, Lincoln, NB.

Figures 6a and 6b show the inner and outer views of the door. The forward hinges, which are installed in the field, are shown in Figure 6c, which shows the door installed on the helicopter.

The door is 37.5 in. long by 23.4 in. wide. Both the inner and outer skins are made from two plies of 120 style Kevlar/epoxy fabric, and one ply of 181 style Kevlar/epoxy fabric with the fibers aligned along and perpendicular to the length of the door. The core is 0.38-in.-thick, 3.1-lb/ft^3 Nomex. As shown in Figure 7, the core is scarfed in the vicinity of the door edges, and the inner and outer edges are bonded together, resulting in an edge thickness of 0.04 in.

The hinges are attached to the door by three rivets through the hinge, as shown in Figure 7. The outer two rivets go through the 0.040 skin only, and the inner rivet goes through the entire sandwich structure. The Nomex core is locally reinforced with core-fill where the hinge rivet penetrates it. At the latches, the structure is reinforced with Kevlar/epoxy fabric to accommodate the locally high attachment loads.

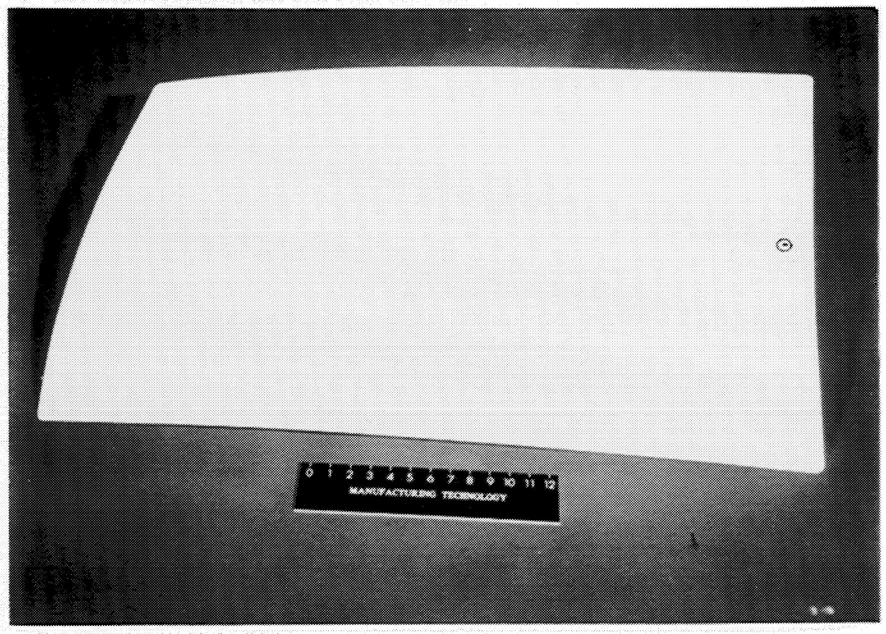
3.4 VERTICAL FIN

The vertical fin is a sandwich structure made from graphite/epoxy facesheets over a honeycomb core. The leading and trailing edges are Kevlar/epoxy and graphite/epoxy, respectively. Figures 8a and 8b are views of the completed fin. An exploded view is shown in Figure 9.

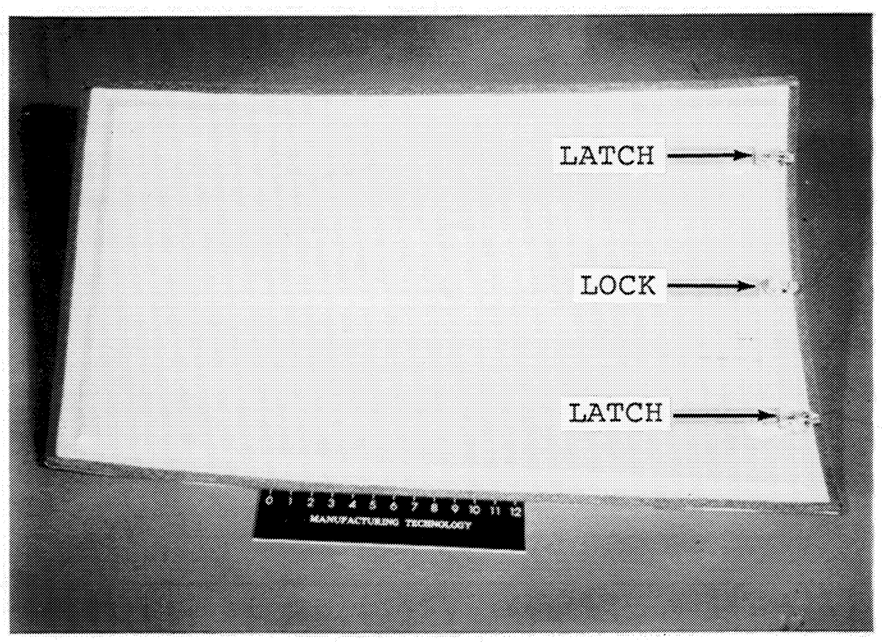
The facesheets are made from T300/788[5] graphite/epoxy whose thickness varies from 0.072 in. in the fuselage attachment region to 0.020 in. near the tip, and are fabricated from spanwise plies of ±22.5 degree tape that runs continuously across the midspan. Additional plies are oriented at +22.5, 67.5, -22.5, and -67.5 degrees, as required, with an overlap in the midspan area. This layup gives essentially [0, ±45,0] properties along the fin's swept structural axis, and isotropic properties in the midspan area. Lightning protection is provided by 200-grid aluminum alloy screens bonded to the outer surface of each face. Insulation between the graphite and the alumimum screen is provided by the adhesive.

The core is made from high-strength fibertruss fiberglass material[6] that has a constant 1.25-in. depth. On the upper fin the

[5]Manufactured by U.S. Polymeric Co., Santa Ana, CA.
[6]Manufactured by the Hexcel Corp., Dublin, CA.

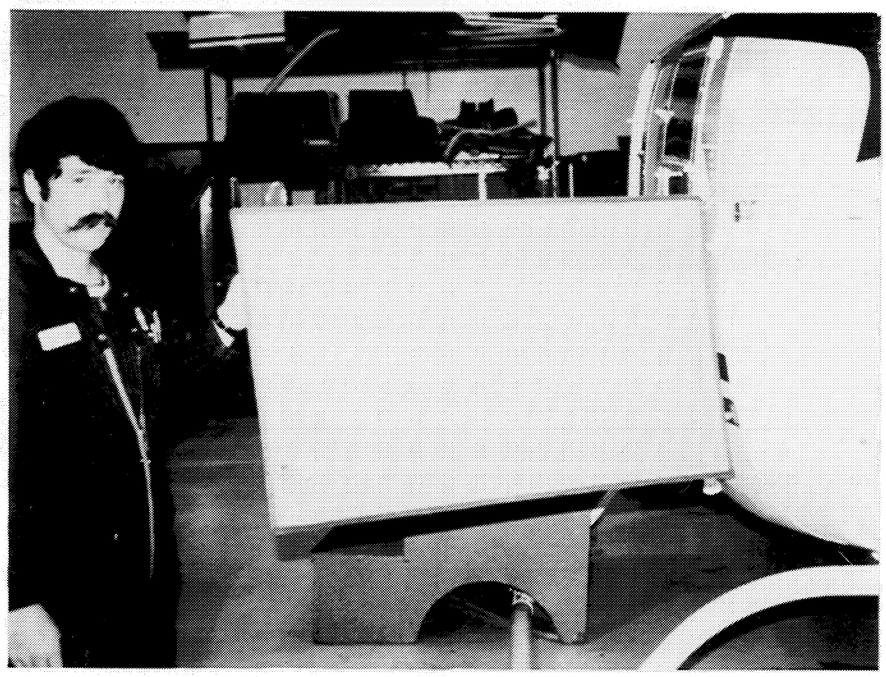


a. Outer view



b. Inner view

Figure 6. Composite baggage door.



c. Baggage door in open position

Figure 6. Composite baggage door (continued).

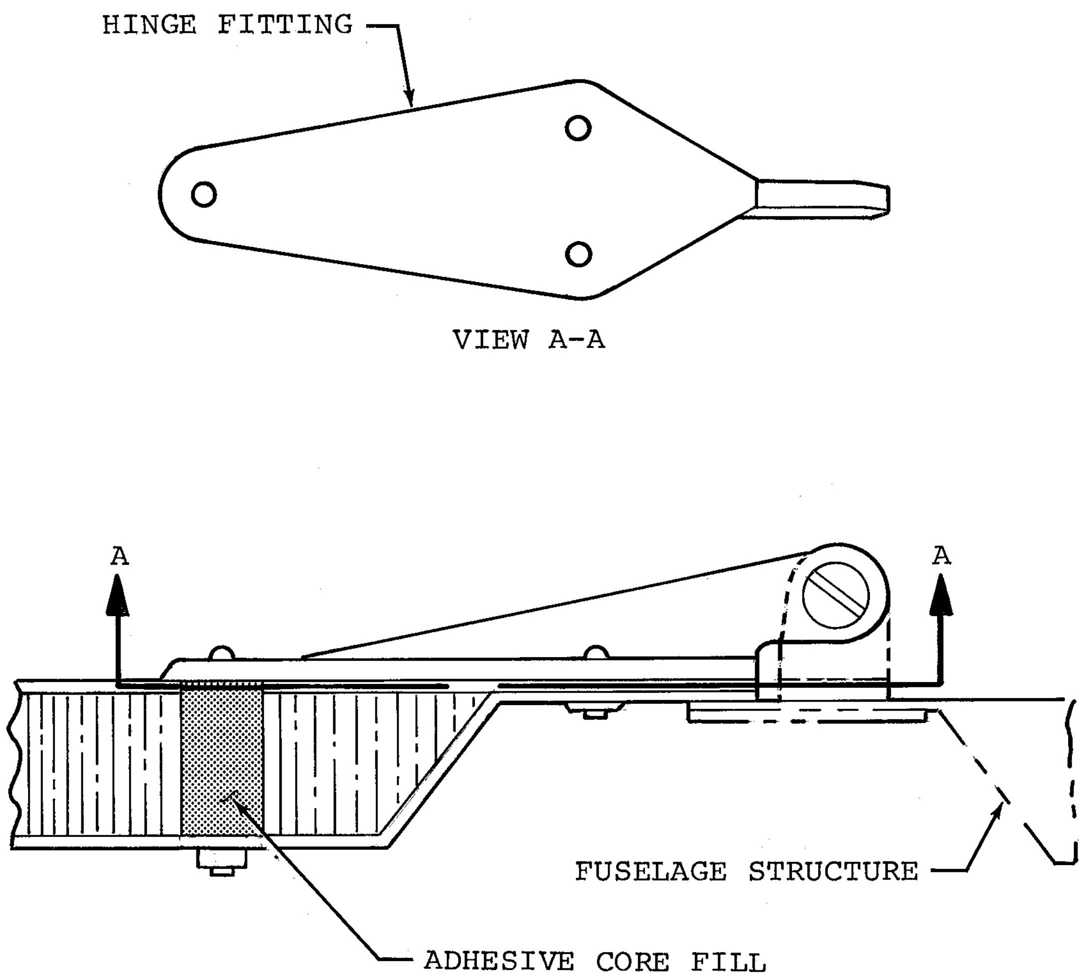
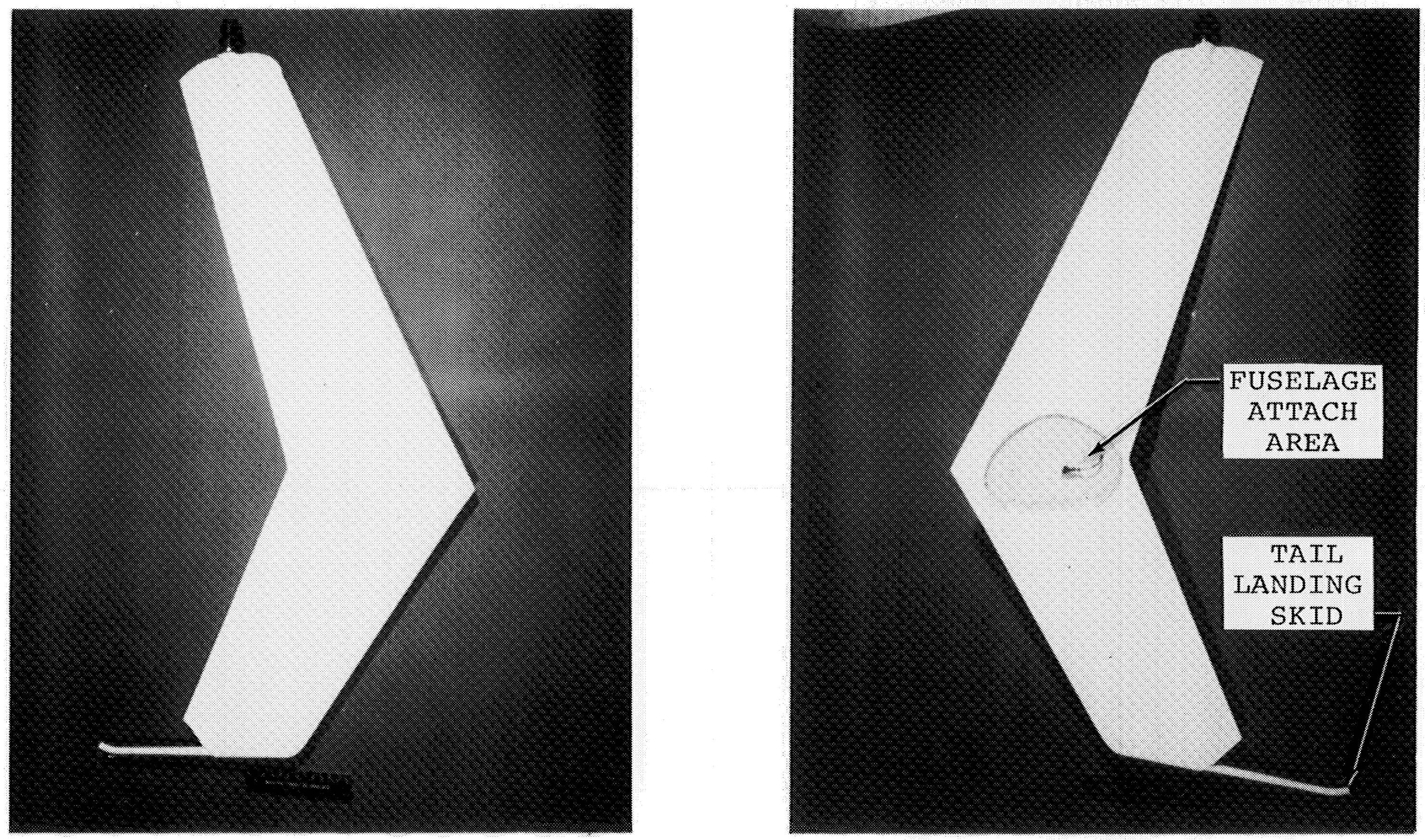


Figure 7. Edge of baggage door, showing hinge attachment.



a. Outer view

b. Inner view showing attachment to fuselage

Figure 8. Graphite/epoxy vertical fin.

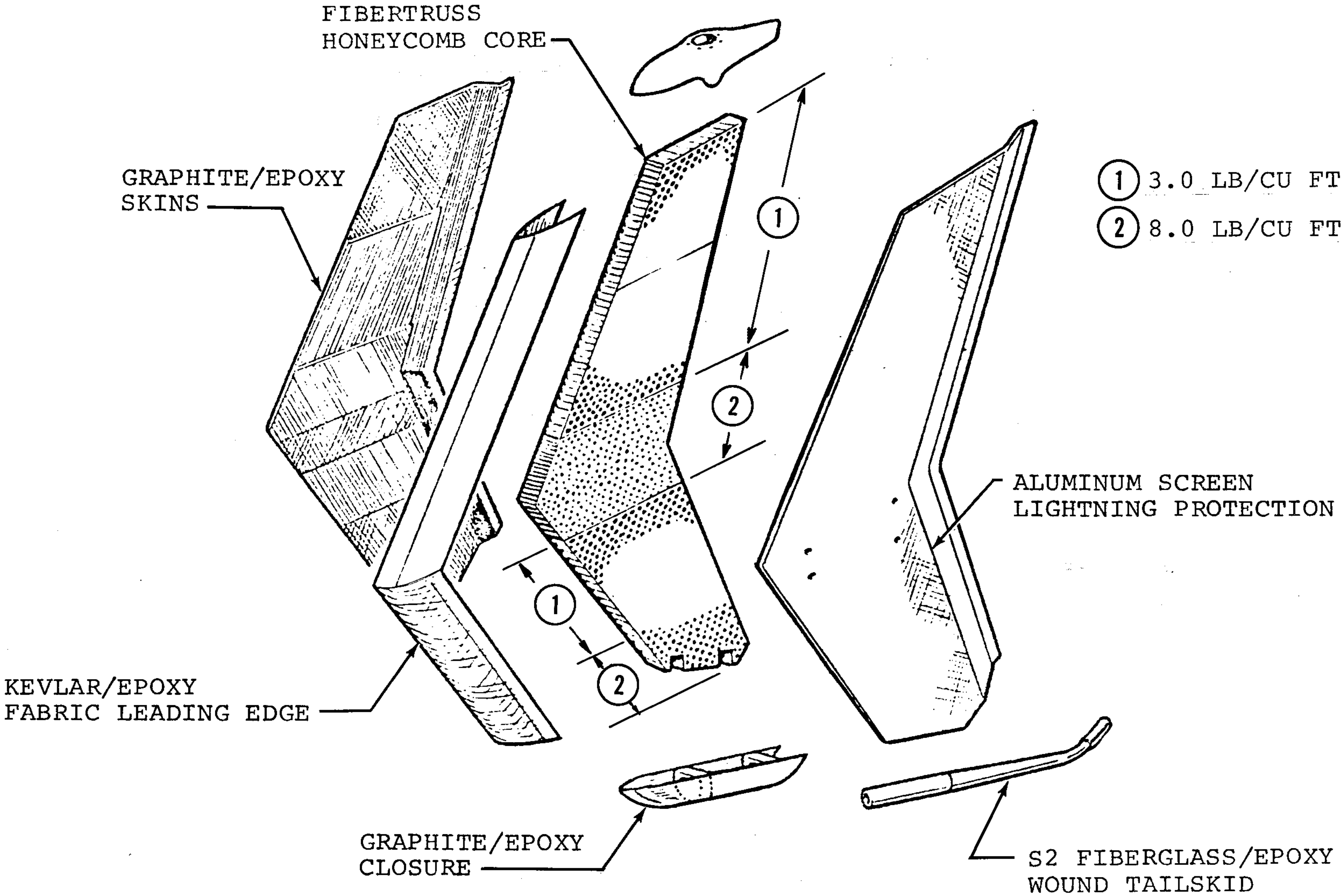


Figure 9. Exploded view of vertical fin.

core has a 3.0-lb/ft^3 density from the tip to the vicinity of the fuselage attachment area, where it changes to an 8.0-lb/ft^3 density. On the lower fin the core has an 8.0-lb/ft^3 density from the tip upward for 12.2 in. where it changes to a 3.0 lb/ft^3 density. This continues upward until it splices to the 8.0-lb/ft^3 core in the fuselage attachment area.

The leading edge is made from two plies of 281 style Kevlar/epoxy fabric. The trailing edge is an extension of the face skins, and it is formed to the aft contour of the fin and bonded together at the extreme trailing edge.

A tail landing skid made from filament-wound S-glass prepreg roving is located at the lower fin tip. The skid is installed into the fin through two molded fiberglass/epoxy[7] blocks, as shown in Figure 9. The blocks are covered by a graphite/epoxy protective shell.

The fin is installed onto the fuselage by four bolts that pass through steel inserts that extend the full depth of the fin and are potted into the structure. This local area is reinforced by graphite/epoxy pads that help distribute the loads from the fin structure to the inserts.

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[7]Material is manufactured by the Fiberite Corp., Winona, MN.

4. ENVIRONMENTAL EXPOSURE SPECIMENS

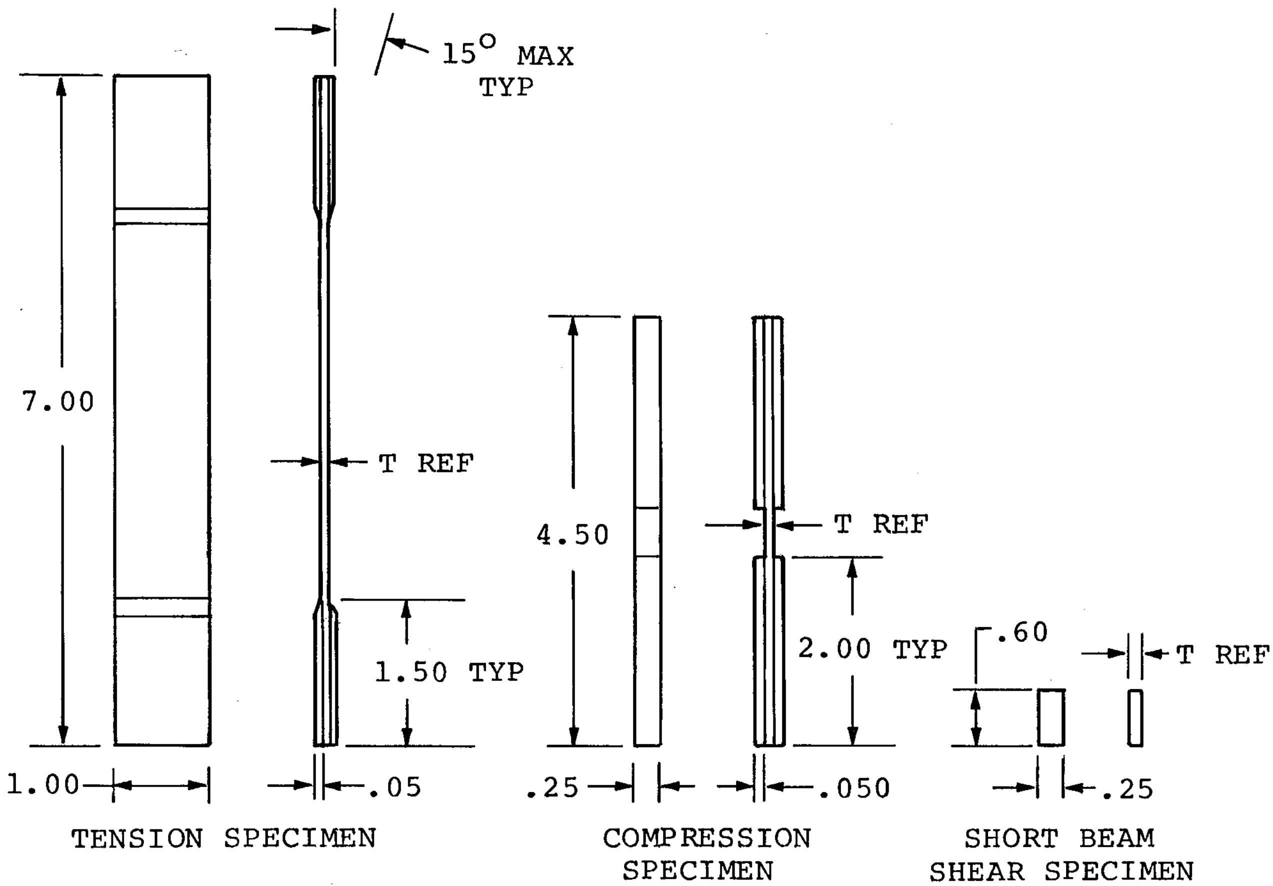
To supplement the flight service data on the composite components, data on environmentally exposed test coupons are being obtained. These coupons are made from the same materials and have the same ply layups as the flight components. They are geographically located in the same environments as the flight service components.

The test specimens are shown in Figure 10. Six hundred and six each of the tension, compression, and short beam shear coupons, and 25 of the 2.00 by 7.00 flat panels were made from laminates representing the external skins of each of the four components. Six of each type of coupon are to be used as control (unexposed) specimens, while the others were installed on specially designed racks for long-term environmental exposure. The racks are located on an oil platform in the Gulf of Mexico, at Cameron, La., NASA Langley, Hampton, Va., Toronto, Canada, and Fort Greeley, about 50 miles southeast of Fairbanks, Alaska.

A typical exposure rack is shown in Figure 11. The racks are made from aluminum alloy and are designed to be stable in a 75-knot broadside wind. The rack located at Fort Greeley has its legs embedded in concrete in the ground because their winds often exceed 75 knots.

Each rack contains five removable panels and each panel contains 76 specimens - 19 for each of the four components: six tension, six compression, six short beam shear, and one unpainted. Each rack, therefore, contains 380 specimens. Figure 11 shows that each panel may be removed from the rack by removing four bolts.

After the first and third years of exposure, 1 panel (one-fifth of the specimens) will be removed from each exposure rack and shipped to NASA LRC Structures Laboratory in moisture-proof containers. After the fifth year, 60 percent of the specimens will still be on the exposure racks. At that time, at the discretion of NASA, the remaining specimens will be tested, or they may remain on the racks for additional exposure before being returned for test.



COMPONENT	MATERIAL	LAMINATE	THICKNESS "T"
BAGGAGE DOOR	KEVLAR 49/EPOXY 120 SYTLE CLOTH	[0/90/±45]	.060-.080
VERTICAL FIN	T300/788 GR/E TAPE	[0/+45/-45/0]	.062-.082
LITTER DOOR	KEVLAR 49/EPOXY 281 STYLE CLOTH	[0/45/0]	.070-.090
FORWARD FAIRING	KEVLAR 49/EPOXY 281 STYLE CLOTH	PLIES [0/90]	.060-.080

Figure 10. Environmental exposure specimens.

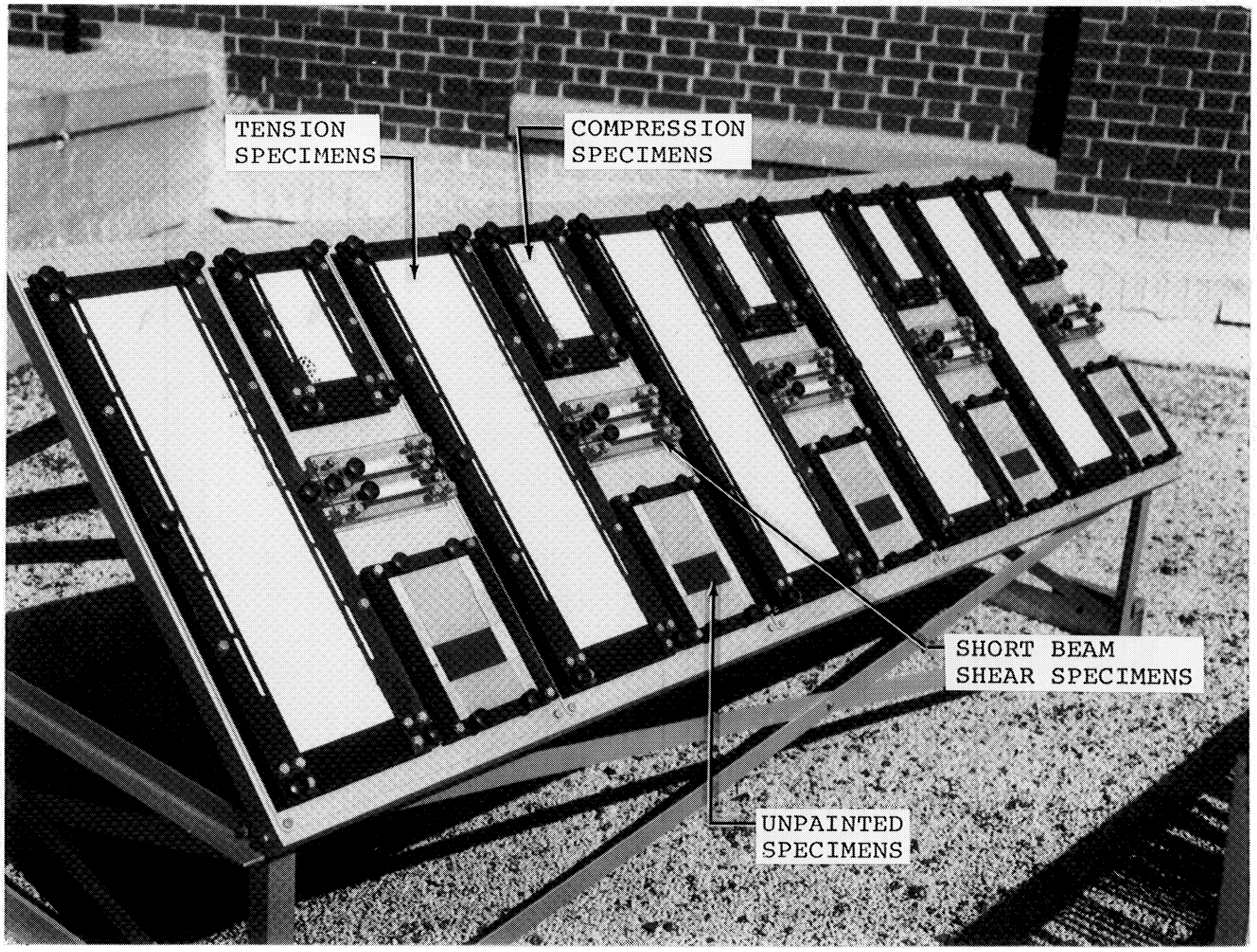


Figure 11. Environmental exposure rack with specimens installed.

5. MANUFACTURING PROGRAM

The Manufacturing program consisted of two phases. The first phase was tool design and fabrication, followed by fabrication of the components for FAA certification. The second phase was the fabrication of 45 of each component using production procedures. Although these are two distinct phases, there is an overlap between the two in that tooling and fabrication concepts had to be developed to produce the components in a production manner. Also, since the components received FAA certification and conformity inspection, all the components had to be fabricated in the same manner and be subjected to the same inspection procedures. A further factor that had to be considered was that BHTI instituted a cost-tracking procedure as part of this program, and any significant changes to the established fabrication procedures were not permitted.

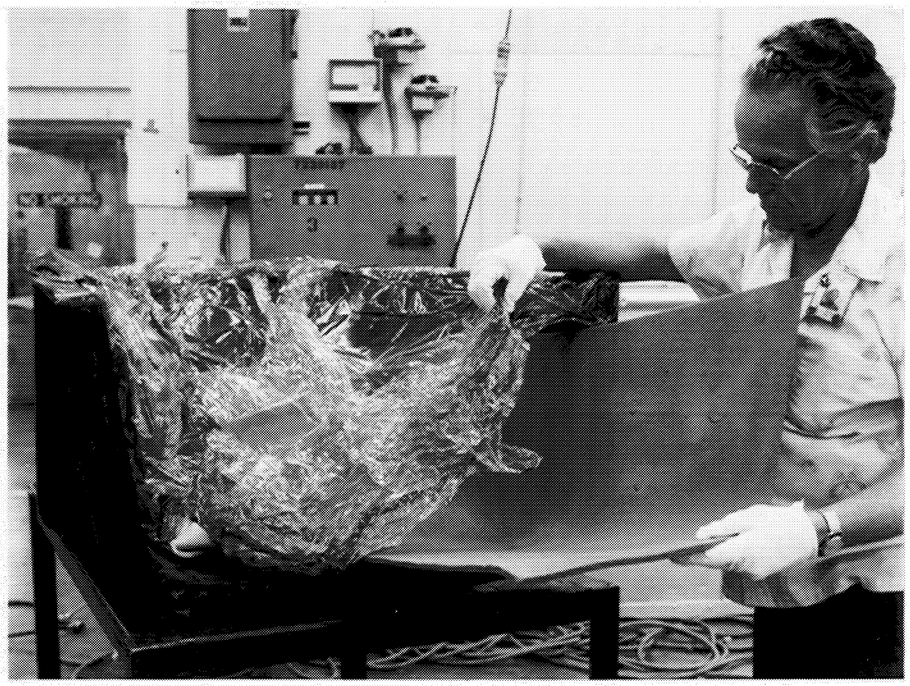
5.1 FABRICATION OF FORWARD FAIRING

As described in Section 3.1, the fairing is a sandwich structure made from a single ply of Kevlar/epoxy fabric for each facesheet, and a Klegecell foam core. A feature of the fabrication process is that the facesheets and core are cocured in one operation, making use of the resin in the prepreg to make the bond.

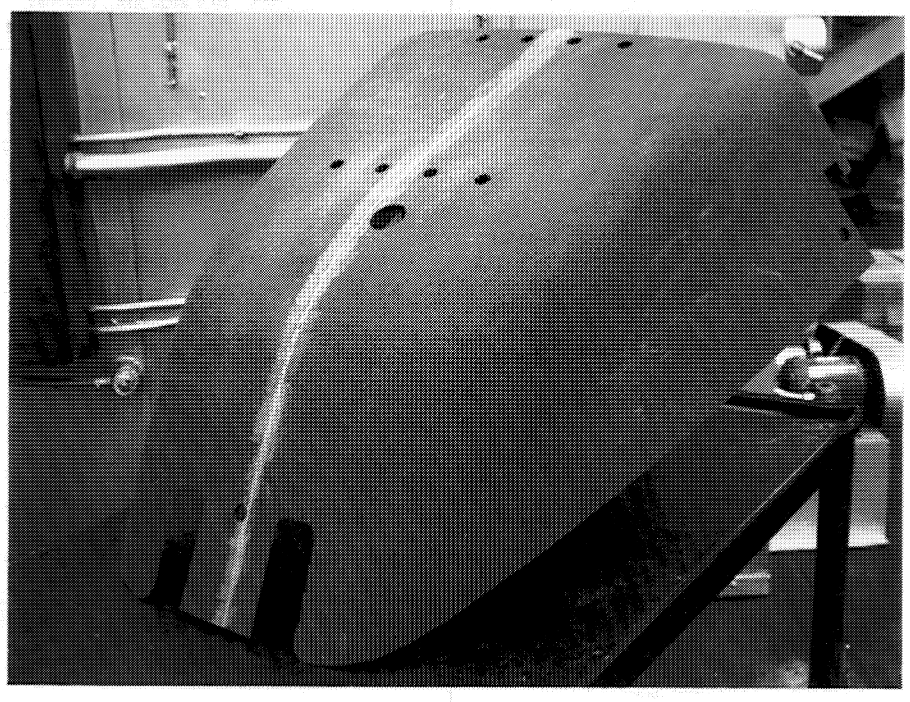
To preform the Klegecell core, the core, a thermoplastic foam, was first heated to 350°F for 2 minutes to soften it. It is then placed in a mold, bagged, and air-evacuated to about 2 in. H_2O. It is replaced in the oven for 5 minutes, during which time a full vacuum is drawn. It is then allowed to cool to handling temperature and the formed core is removed. Because of the severe double curvature near the forward end, the core was made in two halves and spliced together along the longitudinal centerline. Figure 12a shows half of the core being removed from the mold, and Figure 12b shows the final core after the two halves are spliced.

Each facesheet is made from three individual sheets that are spliced together longitudinally at the corners. Corner splices were required to permit the skins to move and conform to the contour of the core under pressure. Figure 13a shows a portion of the inner facesheet in the mold prior to the cure cycle, and Figure 13b shows an unpainted fairing. The longitudinal splices can be seen in the figure. The cure cycle is performed at 200°F for five hours.

Figure 2a and 2b show a completed forward fairing minus the hinge, which is added in the field.

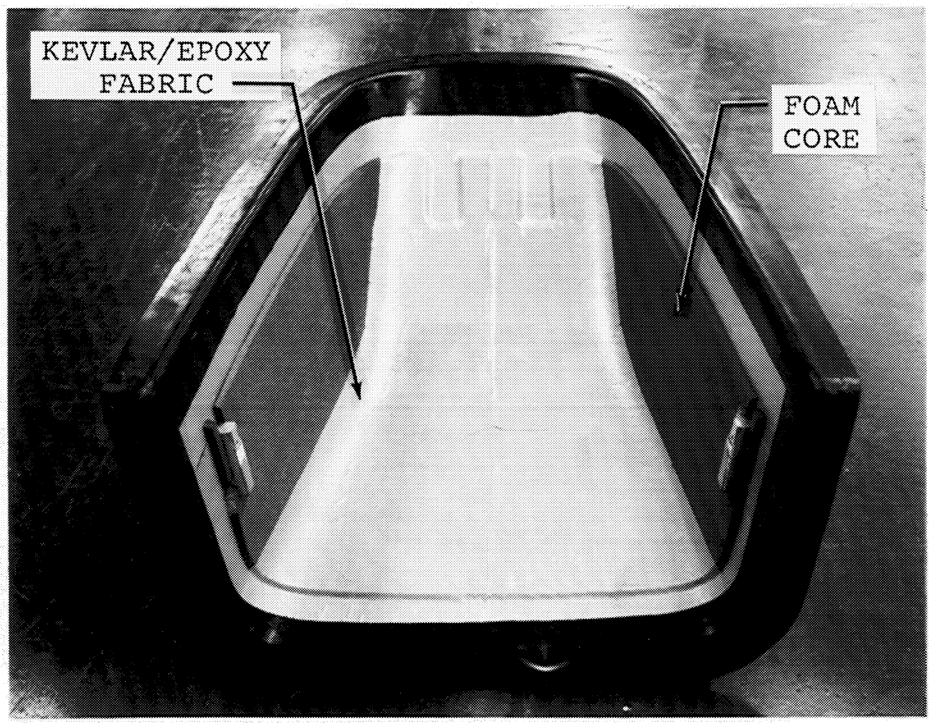


a. Core half being removed from mold

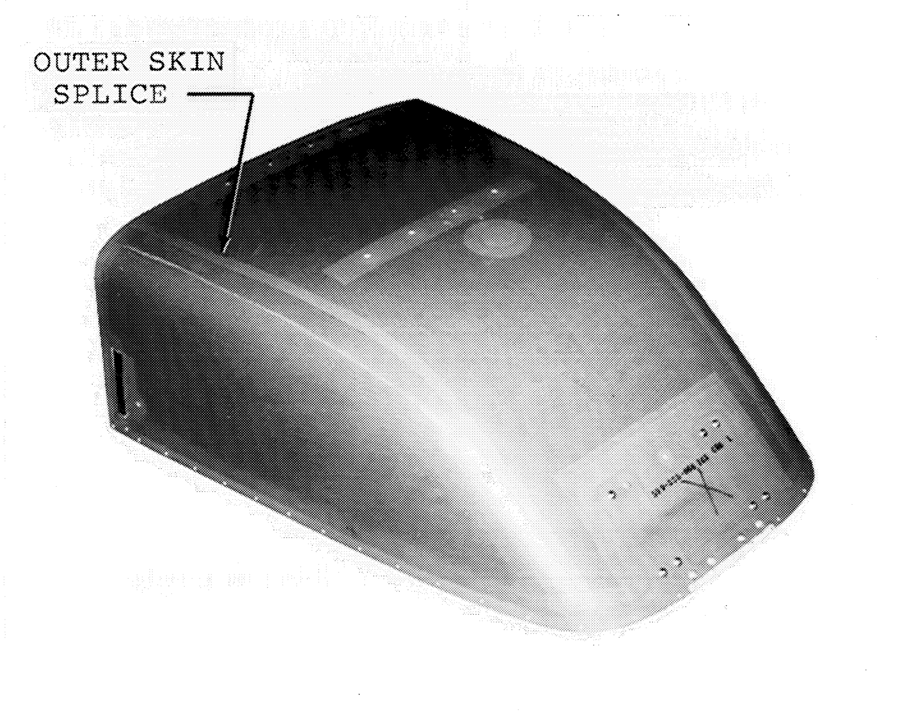


b. Completed foam core

Figure 12. Fabrication of foam core for forward fairing.



a. Core and portion of inner skin in mold



b. Unpainted fairing showing longitudinal splices

Figure 13. Forward fairing construction using longitudinal splices.

The Kevlar/epoxy used for the faces has a recommended cure of 260°F for 1.5 hours. However, the fairing was cured at 200°F for 5 hours. Initially, the cocure was tried at 260°F, but it was found that if the Klegecell was exposed to temperatures over 200°F for any significant length of time it would partially collapse. Because cocuring was desired, the temperature was reduced to 200°F for a longer period of time. A 260°F postcure was tried, but even with no pressure the 0.38-in. Klegecell collapsed by about 0.06 in. The degree of collapse was not uniform, nor was it repeatable, and could not be accounted for in design. Therefore, postcuring was abandoned.

There was some concern that the low cure temperature would cause excessive moisture absorption or lower-than-acceptable elevated-temperature properties. The test program that was performed to evaluate these potential problems is described in Section 6.2.3.1.

5.2 FABRICATION OF LITTER DOOR

Figure 14 is a schematic of the "one shot" fabrication technique as it was initially developed. A two-part closed-cavity Kevlar/ epoxy tool is used with each skin. The cavity in the part is vented through the tool to allow autoclave pressure into the cavity. This can be accomplished by either a hole through the part or by a tube. The tool is then envelope-bagged and the part cured in an autoclave. The autoclave air pressure in the cavity presses the part against the tool surface in the same way that an internal bag would. A series of small bleed holes collects air leakage, then passes it through a series of collector grooves into the vacuum bag.

The internal metal hardware is installed after the bond assembly is completed and trimmed, and the plexiglass window is bonded to the door. The hinges are installed in the field because each door is custom-fitted to both the fuselage structure and the passenger door.

Several precertification development bond assemblies were made in the above manner and, although several good parts were made, a change was made in the fabrication process. The two halves were separately bagged and cured, then bonded together with Narmco 1113 adhesive[8]. One of the major problems with the original concept was that loose resin flowed through the vent holes and bonded the tool to the door. It then became a time-consuming task to separate the two, despite the use of a release agent.

[8]Manufactured by Narmco Division of Celenese Corporation, Costa Mesa, CA.

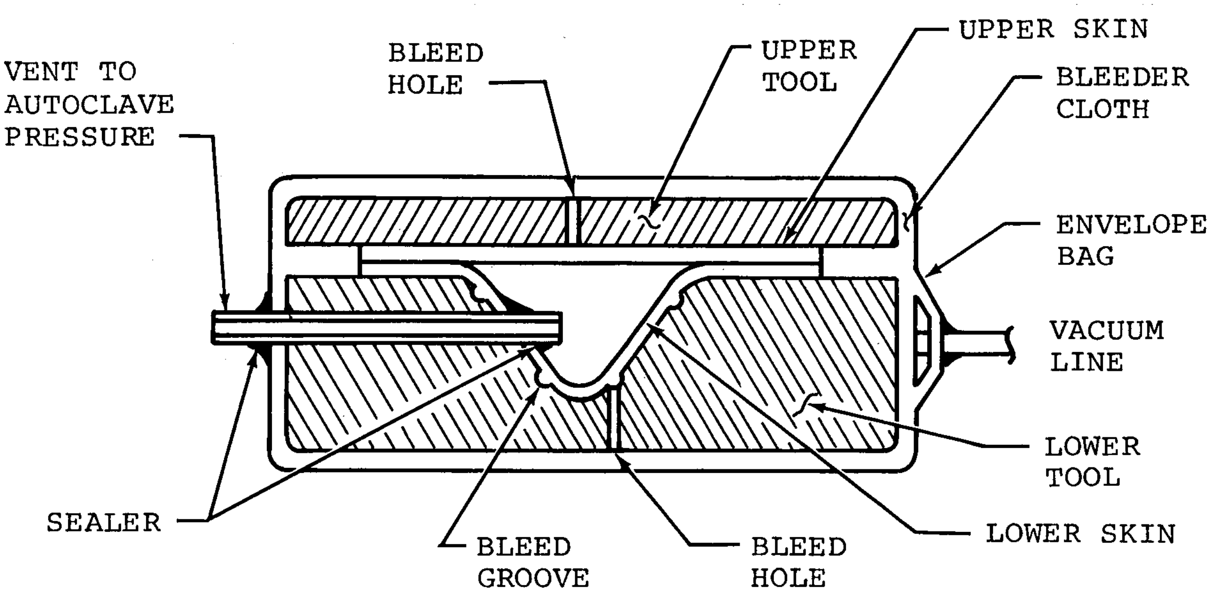


Figure 14. Schematic of "one shot" process as initially designed for the litter door.

It is quite probable that with further development the original concept could be developed and would be a viable production concept. However, in the interest of expediency, the two halves were separately cured and bonded together.

During the course of the manufacturing program, a problem arose that was not discovered until most of the doors were made. Figure 15a shows a door half in its bond fixture under a partial vacuum. It can be seen that the top and bottom edges of the fixture are not rigidly supported all along the length. As a result, the fixture bowed and took a permanent set, causing a bow in the upper and lower edges of the door. This was apparently a gradual phenomenon because the first door had little or no bow, but the bow was very pronounced in the later doors. Figure 15b shows a typical bow in one of the later doors.

In an attempt to correct the problem, a jig was built that had the same contour as the door, and the door was clamped into it as shown in Figure 16a. (Note the heavy bar along the edge to hold it to the jig.) The assembly was placed in an oven and heated to 220°F for 4 hours and then allowed to air cool. Figure 16b shows that the technique was successful in returning the edges to a straight line.

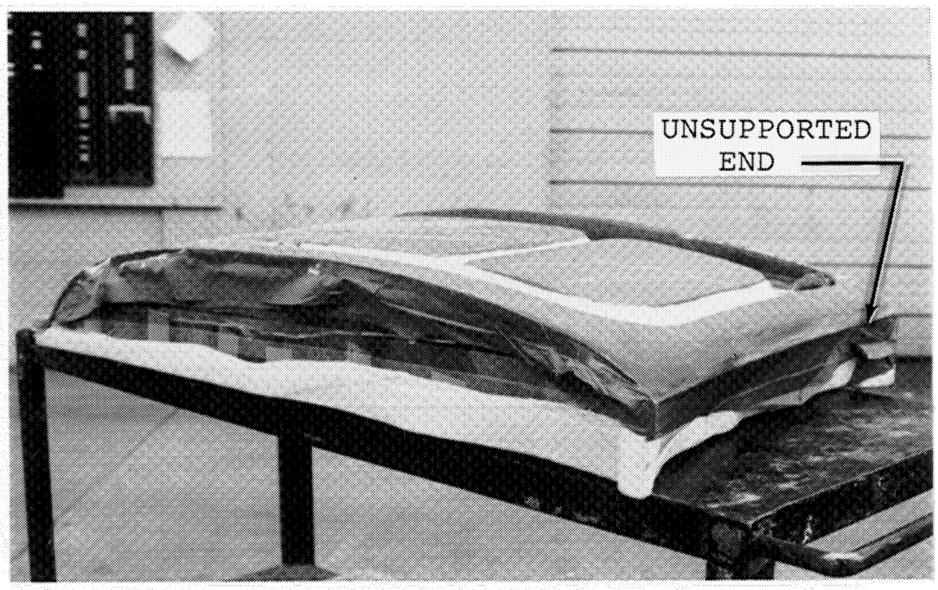
It was necessary to remove the plexiglass window before the door was straightened because of the large difference in thermal expansion between the plexiglass and the Kevlar/epoxy. After the straightening process, the window was replaced.

5.3 FABRICATION OF THE BAGGAGE DOOR

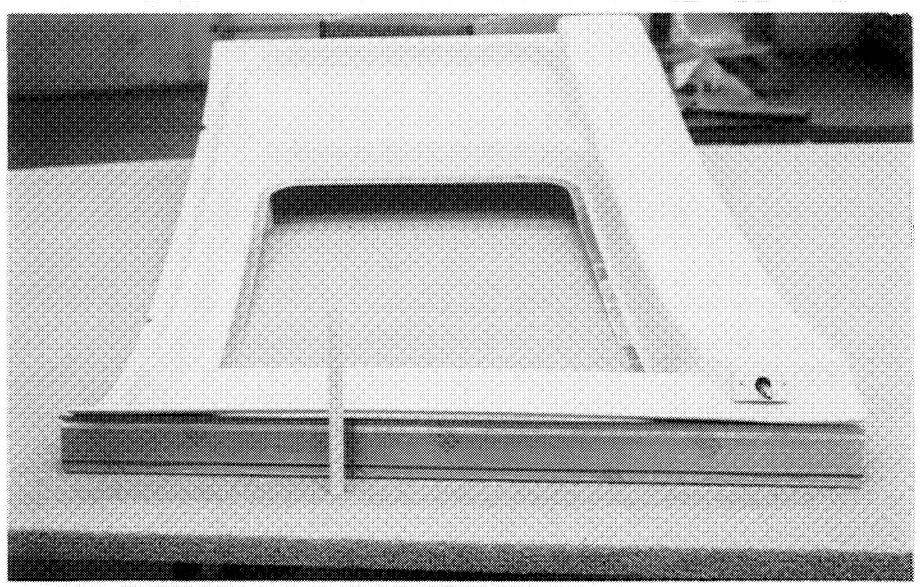
The baggage doors were built by the Brunswick Defense Division of Lincoln, Nebraska. They were fabricated by a conventional hand layup of Kevlar/epoxy fabric over a Nomex honeycomb core. The tool was made from fiberglass/epoxy and was designed for a limited production run.

Brunswick is a company that specializes in wet layups and wet windings; therefore, they used their own proprietary resin system to impregnate dry Kevlar cloth to make prepreg fabric. This fabric was first laid in the mold, the core laid over it, and the inner faces laid over the core. An adhesive was used between the core and the faces. The assembly was then bagged and cured in an autoclave at a temperature of 250°F and a pressure of 40-50 psi. Following the cure, the part was trimmed and clean-cut holes for the lock and latches were made with a water jet cutter.

The latches were installed at Brunswick and the locks at BHTI. As with the fairing and litter door, the hinges are installed in the field to match the mating hinge halves.

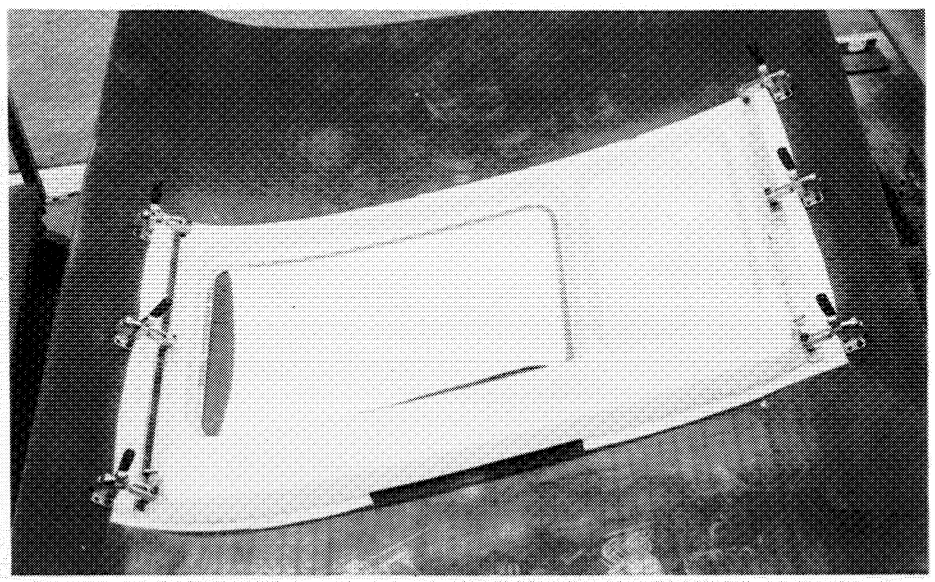


a. View of door half in fixture.

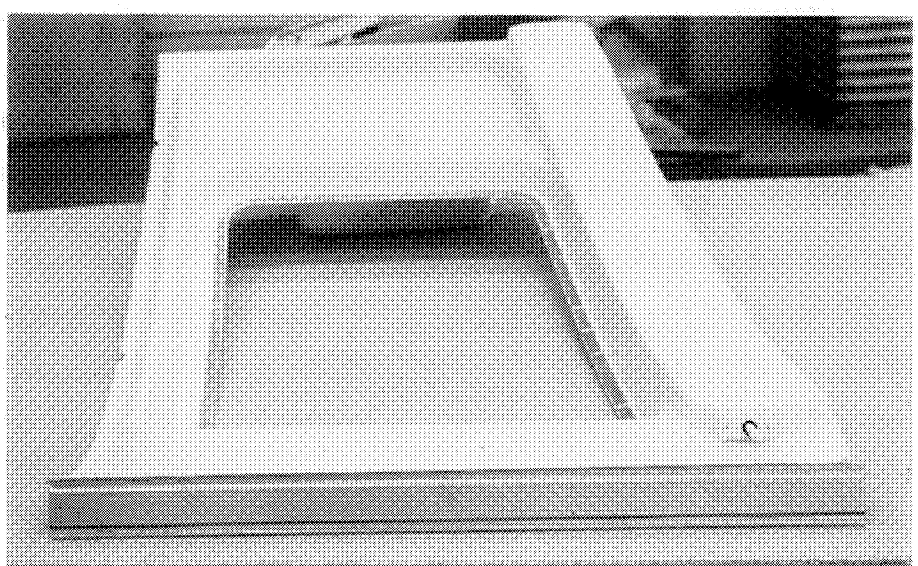


b. Bow of edge.

Figure 15. Litter door bow caused by flexibility of fixture.



a. Door in straightening fixture.



b. Litter door after straightening procedure.

Figure 16. Straightening of bow in composite litter door.

5.4 FABRICATION OF THE VERTICAL FIN

The tool for fabrication of the vertical fin is the main assembly cavity mold. Figure 17 shows the mold being machined. This tool was designed to serve two purposes: to fabricate the precured skins, and to position and secure all details during the final bonding operation.

At the onset of the tooling program, a choice of materials had to be made for the cavity mold. From the standpoint of thermal compatibility with the fin, graphite/epoxy appeared to be the logical choice. However, since graphite/epoxy tooling is expensive and not as durable as steel, steel was chosen for the cavity mold material. This proved to be a good choice, because the skins gave no evidence of buckling due to thermal mismatch and the tool was in excellent condition after all the fins were completed; nor did it require any maintenance during the fabrication program.

Closed-cavity mold assembly tools have proved effective for this type of structure when Nomex core is used, since Nomex exhibits some thermoforming characteristics and, when cut slightly oversized, will conform to the desired dimensions under heat and pressure. However, fibertruss core has negligible heat-forming capability, so a thin silicone liner was added to one face of the mold to provide some tolerance to the tool.

The facesheets are separately cured in one-half of the mold cavity. The first layer placed in the mold is FM1000[9] adhesive followed by the screen wire. The graphite/epoxy prepreg is then laid in by hand and the tool bagged. The layup is then cured in the autoclave at 350°F and 80 psi pressure.

The lower graphite/epoxy closure that houses the tail skid is precured to the shape shown in Figure 9, and bonded to the two molded fiberglass/epoxy fittings with FM53 adhesive[9] to form the closure subassembly shown in the figure. The flat fibertruss core is precut and spliced to form one subassembly. All of the subassemblies are placed in the mold; FM53[9] adhesive is used to bond the facesheets to the core and lower closure. The mold is then closed and the assembly is cured for 90 minutes at 270°F.

After completion of the bond assembly, the filament-wound tail skid, fin-to-fuselage fairing angles, electrical wiring, and upper light are installed. A detailed description of the fin fabrication process is given in Reference 1.

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[9]Manufactured by American Cyanamid Corp., Havre de Gras, MD.

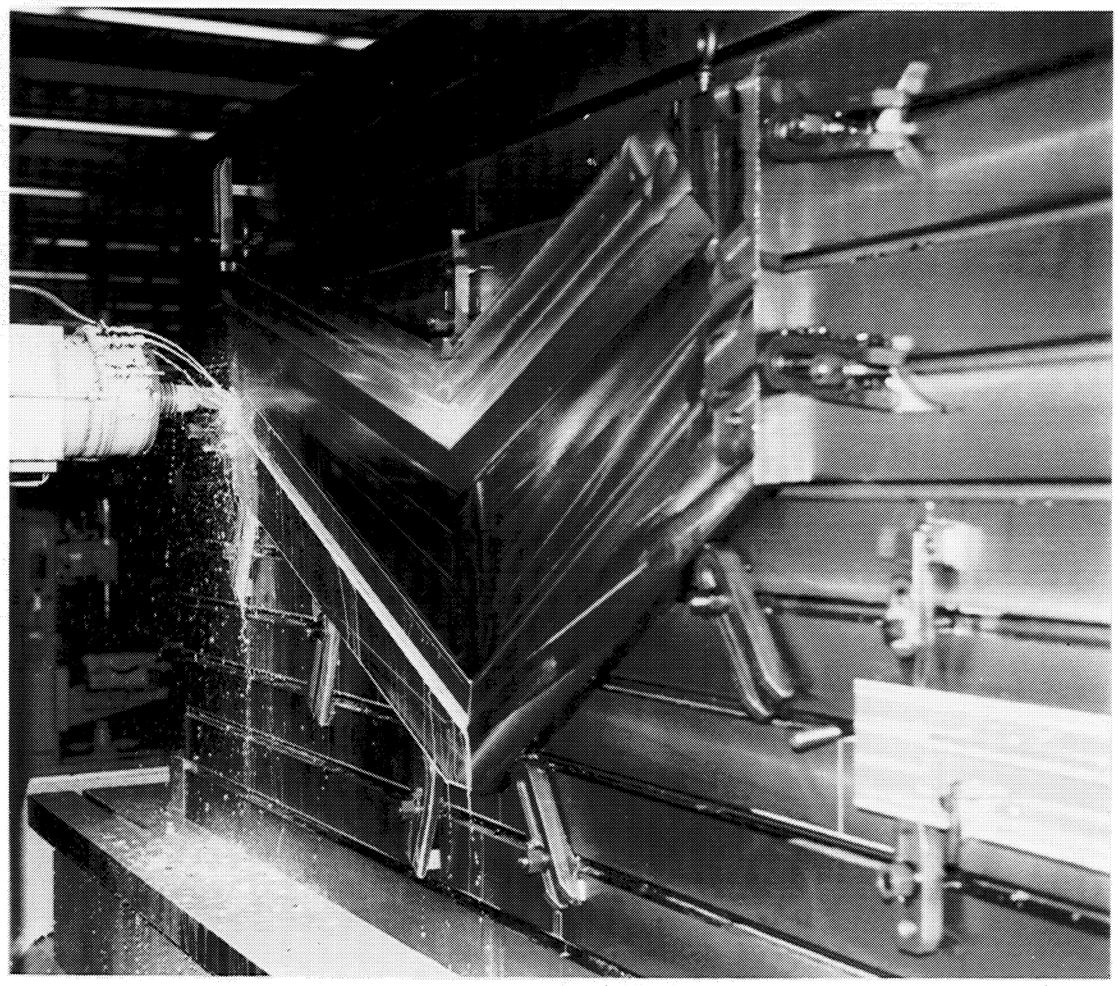

Figure 17. Mold for fabricating graphite/epoxy vertical fin.

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5.5 PAINT, FILL-AND-FAIR

When estimating the potential weight saving that can be realized from advanced composite materials, the weight of the paint, fill-and-fair, is often neglected. This can sometimes lead to unduly optimistic weight estimates, especially if the structure is thin skinned, and if contour is difficult to maintain. The phenomenon is discussed below.

Since the composite components are installed on aircraft that are sold in the competitive commercial market, it is necessary that their surface finish be of a high quality. This means that the painted surfaces must be smooth, and that any irregularities in contour must be removed. Often the parts will have some contour irregularities after the assembly is removed from the mold. This is especially true of parts that have severe double curvatures, such as the forward fairing. These spots will be filled by the paint shop to obtain a smooth contour. Then, to get a smooth surface on which to apply the paint, a coat of sanding surfacer is applied and sanded until a thin layer remains.

If the composite component has few plies, the weight of the fill, sanding surfacer, and paint can be significant and must be considered when deciding whether or not composite materials will be lighter in a specific application.

Table 2 shows the weight distribution based on ten random forward fairings. To get this weight distribution, the bonded assemblies were weighed before and after painting. The hardware and seals were then weighed separately. The difference between the final fairing weight and the individual weights was classified as "miscellaneous". It can be seen that the paint, fill-and-fair is a significant part of the total weight. It is 27.3 percent of the bond assembly weight, which in most designs is as much weight as one can expect to save by the use of composites. It is true that the metal fairing is also painted, so the weight penalty shown is somewhat severe; but the weight of paint is not as large as for the composite fairing.

In a structure that has heavier skins the weight penalty is less because the percentage of weight due to fill, fair, and paint, relative to the structural weight, is less.

TABLE 2. FORWARD FAIRING WEIGHT DISTRIBUTION
AVERAGE OF 10 FAIRINGS

Item	Weight (lb)	Percent Total Weight
Hardware and Seals	1.36	18.7
Bond Assembly	4.40	60.6
Paint, Fill-and-Fair	1.20	16.5
Misc (Bolts, Rivets, etc.)	0.30	4.2
Total	7.26	

5.6 COST TRACKING

The objective of this part of the program was to determine the cost of the composite components as a function of the number of parts made. At the present time there are relatively few composite airframe parts in production on American helicopters, so this information can help to either verify or refute production cost projections made for similar types of composite structures.

The most important item to be established is the man-hours required to make a specified part. Material costs are undoubtedly important, but composite material costs are continually changing.

The amount of material used on a part can be estimated by studying the drawing and adding a factor for scrappage; so the material cost can be computed at any specific time by using the appropriate unit cost.

The cost of tooling was not amortized in this study because some tools, like the ones for the fin and fairing, could be used for many more parts, while the tools for the two doors barely made the number of parts required for the program.

The man-hours required for each of the three parts made by BHTI are shown as conventional log-log plots. To get these data, each part was serialized and a traveler assigned to it. The traveler specifies each operation required to make the part. As the workers perform each operation, they charge their time to the specific traveler. When all of the work is completed, the traveler is closed out. The Accounting Department then collects and tabulates all of the time charged to each traveler.

Inspection of Figures 18 through 20 shows rather large fluctuations in the number of man-hours required to build the components. This is apparently normal for the production of as few as 40 to 50 parts. Discussions with Industrial Engineering personnel reveal that components of comparable complexities require a production run of approximately 200 to 300 parts before the fluctuation damp out. It is probable that the same is true for the composite components.

Figures 18 and 20 show a trend toward decreased manhours for the fairing and fin. Due to the limited number of parts involved, and the fluctuation in man-hours, learning curves were not shown for the components. However, the data are presented in a form from which they may be readily computed.

Figure 19 shows an increase in man-hours with increasing number of litter doors. If, however, the first four doors are ignored, then the man-hours decrease slightly with increasing number of doors. An explanation for this is that the first four doors were used for FAA certification tests, and were used before the bows were found in the doors.

The other 46 doors were straightened (Par. 5.2). The man-hours for the straightening process, which included removing and replacing the window, was charged to each of the 46 doors, but not to the first four.

It is pointed out that at BHTI, Quality Control and Painting are defined as Process Labor, which is carried as an overhead function. The charges, therefore, are not shown in the man-hours per part, but are prorated in the same manner as any other overhead charge.

The data on the baggage doors, supplied to BHTI by the Brunswick Defense Division, were not in a form in which curve comparable to those of Figures 18 through 20 could be developed. The cost data provided by Brunswick are shown in Table 3. Here it can be seen that Quality Control, Finishing (Painting), and Ancilliary Labor are not classified as overhead functions at Brunswick, but as direct charges. The data shown in Table 3 are quite detailed and, if a curve were drawn through the first 30 units, it would be steep, and then reverse itself.

The cost of the final 15 baggage doors points out an important fact. After Brunswick completed the 30th door, they were ahead

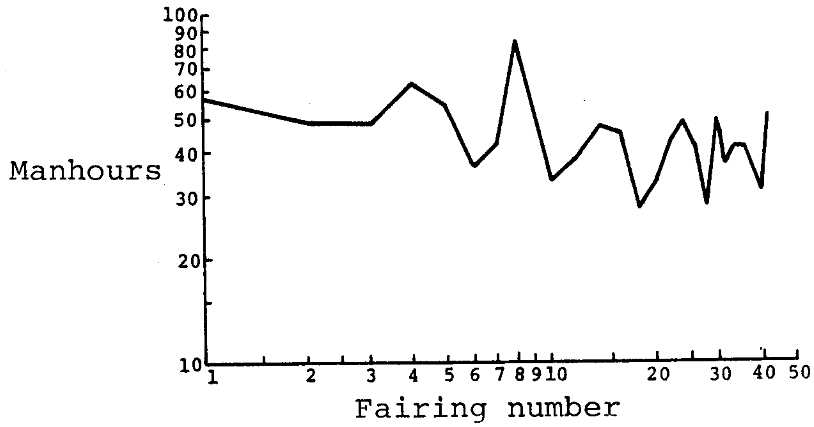


Figure 18. Manhours required to fabricate forward fairing.

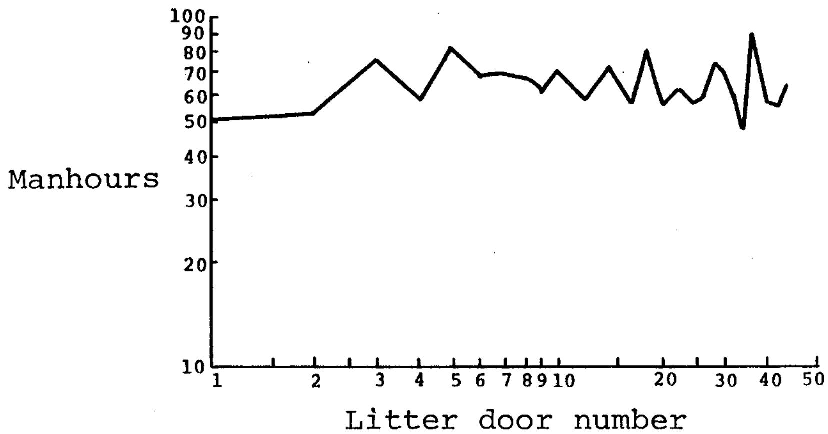


Figure 19. Manhours required to fabricate litter door.

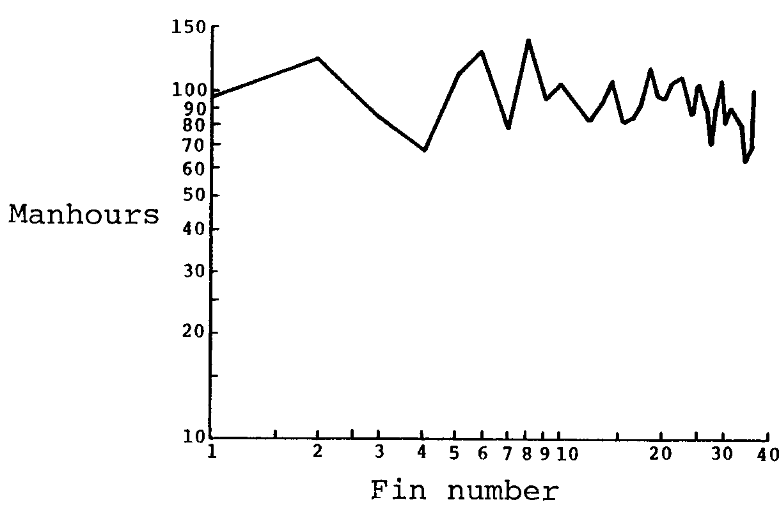


Figure 20. Manhours required to fabricate vertical fin.

TABLE 3. MAN-HOURS REQUIRED TO FABRICATE BAGGAGE DOOR

Unit		Layup Cost	Curing Cost	Finish Cost	Quality Control Cost	Ancillary Cost	Total Cost	Per Unit Shipped
Shipped	Scrapped							
10	11	357	54	193	189	102	895	89.5
5	1	112	17	61	72	28	290	58.0
5	1	103	16	56	60	28	263	52.6
5		66	10	36	14	25	151	30.2
5		57.5	9	31	16	13	126.5	25.3
*5		90	14	49	50.5	18	221.5	44.3
*5	1	112	17	61	74	21	285	57.0
*5	1	128	19	70	74	19	310	62.0
Total 45	15	1025.5	156	557	549.5	254	2542	

*Cost after restart with new personnel and more stringent finish requirements (see text).

of schedule, but there was a question concerning the quality of their paint finish. At that time they stopped work on the doors until the paint problem was resolved. When they resumed production, it was with new personnel, but under the same supervision. It can be seen that, although the Finish cost rose somewhat, (because of more stringent requirements), the layup, curing, and Quality Control costs were more than doubled.

The cost experience of the baggage door points out that today the fabrication of composites is still essentially a craft, and is not automated to the degree that the work can be given to a new group of workers (albeit under the same supervision) without a sharp rise in the cost of the product.

5.7 WEIGHT SAVING

The weight comparison between the composite and corresponding metallic components is given in Table 4. The acutal weights of the composite components are the average of ten of each component randomly chosen after they were completed, including paint, all hardware installed, and an allowance for hardware to be installed in the field.

The table shows that there was no weight saved by the composite baggage door. The weight of the door is high because a severe stiffness requirement was placed on the design, and the depth of the door was held to a minimun to obtain the maximum possible baggage volume . A study has shown that the weight can be reduced by removing one ply of fabric from each face and locally stiffening the door with graphite tape.

TABLE 4. WEIGHT COMPARISON OF COMPOSITE COMPONENTS WITH METALLIC PRODUCTION PARTS

Component	Wt of Metal Part (lb)	Wt of Composite Part (lb)	Wt Saving (lb)	Wt Saving (%)
Fwd. Fairing	8.60	7.26	1.34	15.6
Litter Door	13.10	8.20	4.90	37.4
Baggage Door	2.90	2.90	0	0
Vertical Fin	15.30	12.30	3.00	19.6
Total	39.90	30.66	9.24	23.2

6. FAA CERTIFICATION PROGRAM

A fundamental requirement of this program is that the four composite components be certified by the FAA for unlimited operation within the flight envelope of the 206L helicopter. This not only requires FAA conformity inspection and continuing quality survey, but also a series of structural tests to demonstrate the strength of the components. Static tests were conducted to failure on all four components, in addition to a fatigue test on the vertical fin.

In conducting structural tests, two options are available to account for the reduction in strength caused by exposure to elevated temperature and the absorption of moisture. The first option is to perform the tests at elevated temperature after being environmentally conditioned for a specified length of time, temperature, and relative humidity. The second is by use of knockdown factors. The knockdown factors are determined by testing representative preconditioned specimens and taking the ratio of the room temperature (dry) strength to the environmentally conditioned strength. The required component static strength at room temperature is then the design ultimate load times the knockdown factor. For the FAA certification program, the first option was initially taken for the vertical fin, while the second option (knockdown factors) was taken for the other three components. Section 6.1 describes the coupon tests that were performed to develop the knockdown factors. The environmental conditions to which the specimens were preconditioned and the test environments were those agreed upon by the FAA and BHTI.

6.1 COUPON QUALIFICATION TEST PROGRAM

6.1.1 Tests and Test Specimens

Tension, compression, and rail shear tests were performed. Flat coupons were used for tension and rail shear tests, and a sandwich beam test specimen was used for compression tests. The specimens are defined in Figure 21. The materials used for each coupon are specified in Table 5, and the thickness and ply orientations are shown in Tables 6 through 8.

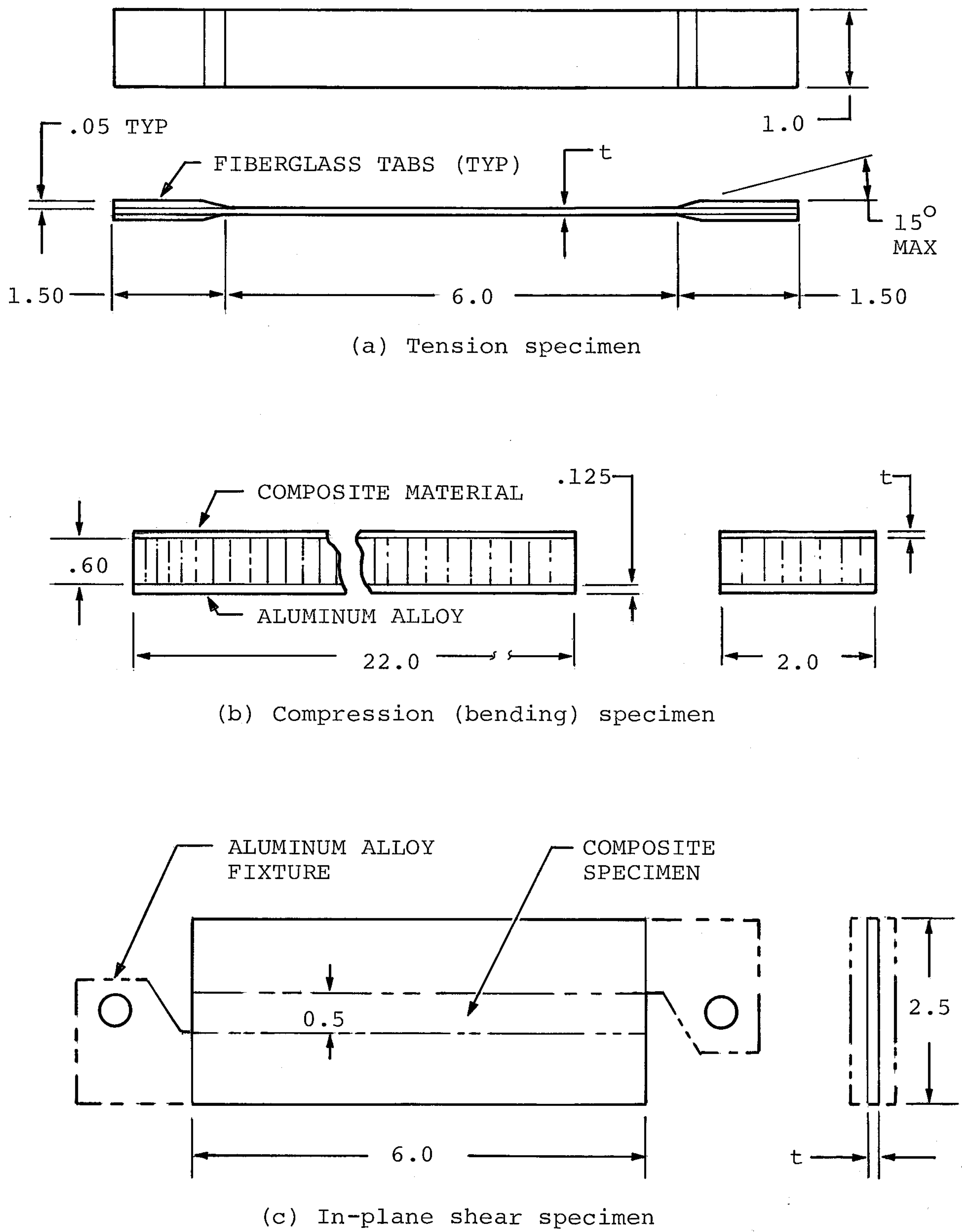


Figure 21. Test coupons used to determine strength loss caused by heat and moisture. (Dimensions shown in inches.)

TABLE 5. MATERIALS FOR TEST COUPON PROGRAM

Component	Fiber Reinforcement	Resin System
Baggage Door	Kevlar-49 120 Style Cloth	Brunswick Corporation Proprietary Epoxy
Litter Door	Kevlar-49 281 Style Cloth	Hexcel Corporation F-185 Epoxy
Forward Fairing	Kevlar-49 281 Style Cloth	Ferro Corporation CE306 Epoxy

TABLE 6. BAGGAGE DOOR TEST COUPON PLY ORIENTATION

	Test Coupon		
	Tension	Compression	Rail Shear
Number of Plies* Ply Orientation Typical Thickness, Inches	6 $(0°/\pm45°/0°)_S$ 0.028	6 $(0°/\pm45°/0°)_S$ 0.028	6 $(0°/\pm45°/0°)_S$ 0.028

*120 Style Kevlar-49 Woven Material

TABLE 7. LITTER DOOR TEST COUPON PLY ORIENTATION

	Test Coupon		
	Tension	Compression	Rail Shear
Number of Plies* Ply Orientation Typical Thickness, Inches	3 $(0°/\pm45°/0°)$ 0.033	3 $(0°/\pm45°/0°)$ 0.033	3 $(0°/\pm45°/0°)$ 0.033

*Kevlar-49 Woven Material.
Outer Ply (0°) 220 Style Woven Material. Two Inner Plies (45°/0°) 281 Style Woven Material.

TABLE 8. FORWARD FAIRING TEST COUPON PLY ORIENTATION

	Test Coupon		
	Tension	Compression	Rail Shear
Number of Plies* Orientation	3 (0°/0°/0°)	3 (0°/0°/0°)	3 Ply (0°/90°/90° 0°)
Typical Thickness, Inches	0.030	0.030	0.040

*Kevlar-49 281 Style Cloth/Ferro Corp. CE306 Epoxy

6.1.2 Specimen Preparation

The laminates with woven Kevlar-49 reinforcement were laid up and cut so that the warp direction was 0°. The compression and tension test coupons were tested only in the warp (0°) direction. The rail shear test specimens were cut with the long axis at 0°.

6.1.3 Test Matrix

Tests, test methods, test temperatures, pretest conditioning of specimens, and number of specimens per test are listed in Table 9.

6.1.4 Coupon Test Results

A summary of the test results is given in Table 10. For purposes of developing knockdown factors, average test data were used. From these data the factors were established and agreed to by the FAA and BHTI for purposes of certifying the composite components.

6.1.5 Knockdown Factors for Components

The knockdown factor is defined as the ratio of the non-conditioned (room temperature dry) failing of the representative laminate to its environmentally conditioned failing stress. Table 10 shows that there should be a different knockdown factor for each mode of failure for each laminate; also, the factor should be applied after the static test, when the mode of failure of the component has been established. However, it was agreed that a "worst condition" knockdown factor would be established for each component, and this should be used regardless of how the component fails. The knockdown factors were therefore conservatively computed on a "worst condition" basis, and are shown for each component in Table 11.

TABLE 9. MATRIX OF COUPON TESTS

Test	No. of Test Coupons*					Total Test Coupons
	-67°F Dry	R.T. Dry	120°F Wet	160°F Dry	180°F Dry	
<u>Tensile</u>						
Baggage Door	3	6	6	3	0	18
Litter Door	3	6	6	3	0	18
Fwd. Fairing	3	6	6	0	3	18
<u>Compression</u>						
Baggage Door	3	6	6	3	0	18
Litter Door	3	6	6	3	0	18
Fwd. Fairing	3	6	6	0	3	18
<u>Rail Shear</u>						
Baggage Door	3	6	6	3	0	18
Litter Door	3	6	6	3	0	18
Fwd. Fairing	3	6	6	0	3	18
Total	27	54	54	18	9	162

*Legend:

Dry - Specimens tested at -67°F, R.T., 160°F and 180°F after stabilization for at least 24 hours in a laboratory environment of approximately 75° ± 5°F and 55 percent relative humidity.

Wet - Specimens tested at 120°F after 42 days exposure to an environment of 125° ± 5°F and 95 ± 5 percent relative humidity.

TABLE 10. SUMMARY OF COUPON TESTS (AVERAGE DATA)

Type of Test	Ultimate Strength (psi) -67°F Dry	R.T Dry	120°F Wet	160°F Dry	180°F Dry
	Baggage Door				
Tension	33,260	32,526	29,996	34,203	
Compression	38,209	29,314	21,125	27,970	
Rail Shear	22,800	18,739	18,552	16,218	
	Litter Door				
Tension	39,817	49,515	49,385	45,596	
Compression	59,544	39,809	25,086	20,471	
Rail Shear	23,087	18,984	10,856	10,356	
	Forward Fairing				
Tension	56,241	67,629	56,832		67,382
Compression	39,792	31,371	26,501		24,711
Rail Shear	11,709	15,203	11,951		9,381

TABLE 11. FAA CERTIFICATION KNOCKDOWN FACTORS FOR TESTS OF THREE COMPONENTS

Component	Knockdown Factor	Condition
Baggage Door	1.39	Comp. 120°F Wet
Litter Door	1.94	Comp. 160°F Dry
Fwd Fairing	1.62	Rail Shear, 180°F Dry

6.2 Component Test Loads and Test Results

The test loads and results for each of the components are summarized in the following section. Static loads for the vertical fin were derived from the requirements of Federal Aircraft Regulation FAR 6. Fin repeated loads and the pressure applied to the fairing and two doors come from 206L flight test measurements. The loads were submitted to the FAA, and approved before the tests were started.

6.2.1 Baggage Door Loads and Tests

Two loading conditions were established for the baggage door: aerodynamic pressure, and a load that simulates a pull at the end of the door while the door is open.

Aerodynamic limit load	=	0.33 psi acting outward
Ultimate load	=	0.33 x 1.5 = 0.50 psi
Required strength	=	Ult. x knockdown factor
	=	0.50 x 1.39 = 0.70 psi
Cantilever limit load	=	25 lb per latch = 50 lb
Ultimate load	=	1.5 x 50 = 75 lb

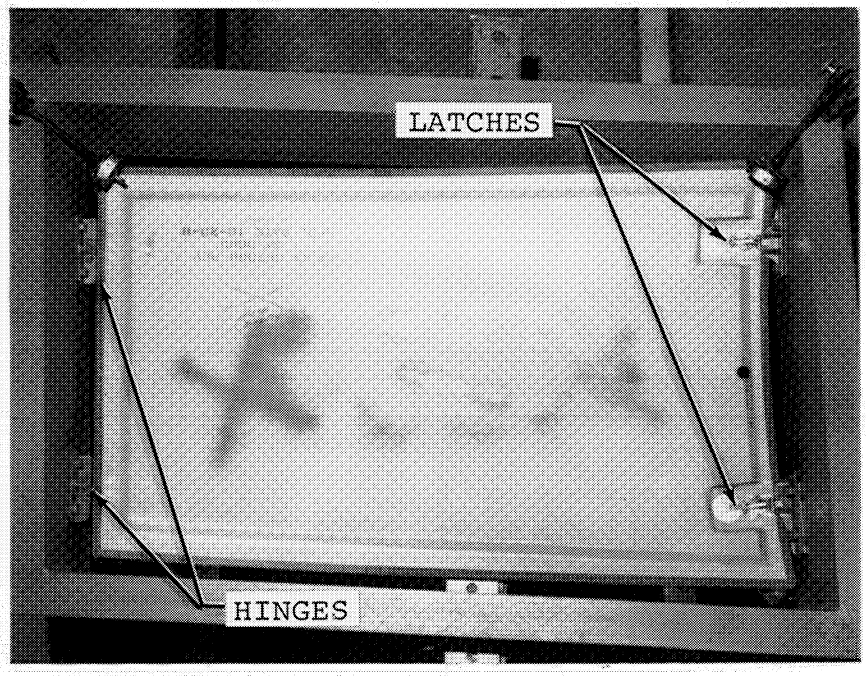
It was only necessary to apply the ultimate load of 75 pounds for the cantilever test. Failing loads were not required. The door suffered no ill effects following application of the 75-pound load.

For the pressure test, the door was tested in the fixture shown in Figure 22a. The door was attached to the jig by two hinges and latches. Pressure was applied by means of water bags. Failure occurred at 0.794 psi, or a total of 695 pounds in the metal hinges, as shown in Figure 22b. This is technically a margin of safety of 13.4 percent, but is based on a metallic hinge failure, not a failure of the composite material.

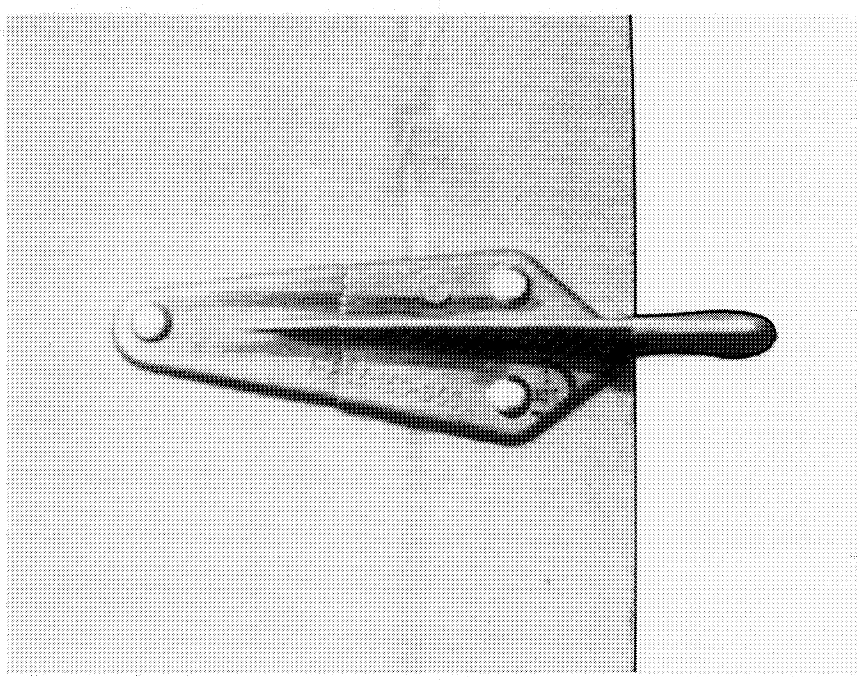
The load-deflection curves of the geometric center of two doors are shown in Figure 23. The curves are nonlinear, so it would appear that there would be a significant permanent set when the load was removed. In the test represented by the x's and dotted lines, the load was removed after 192 pounds had been applied. Despite the nonlinearity of the load-deflection curve, the permanent set was a negligible 0.018 inches.

6.2.2 Litter Door Loads and Tests

As with the baggage door, the litter door was tested for aerodynamic pressure and as a cantilever. The cantilever load was applied through the cabin door when both doors were open.



a. Test setup



b. Static test failure of hinge

Figure 22. Static test of composite baggage door.

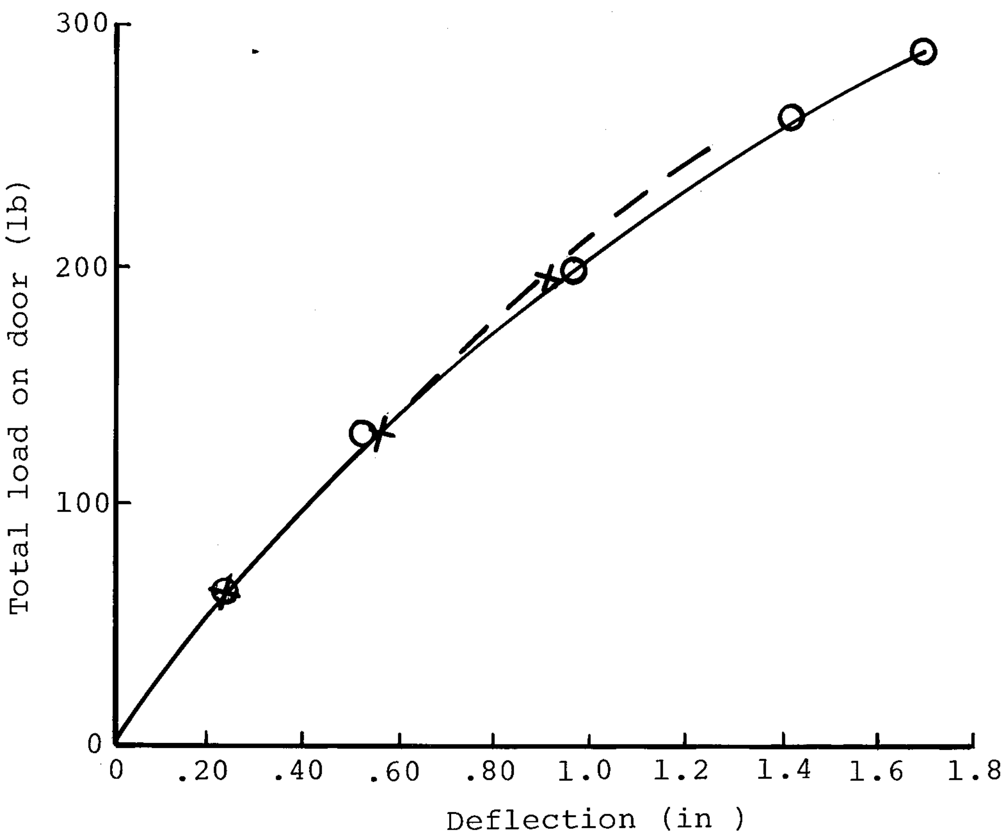


Figure 23. Deflection of geometric center of two baggage doors under test load.

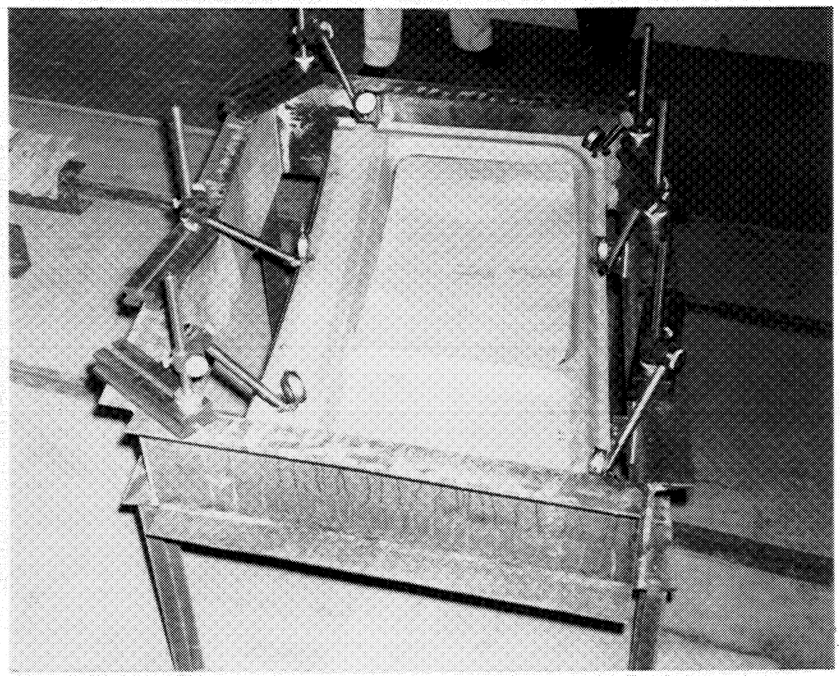
Aerodynamic Limit Load	=	0.20 psi acting outward plus 53 lb at the upper cabin door hinge and 140 lb at the lower.
Ultimate Load	=	0.30 psi plus 79.5 lb at the upper hinge and 210 lb at the lower.
Required Strength	=	Ult x knockdown factor (1.94)
	=	0.58 psi plus 154 lb on the upper hinge, and 407 lb on the lower.
Cantilever Limit Load	=	50 lb at cabin door handle
Ultimate Load	=	75 lb
Required Strength	=	75 x 1.94 = 146 lb

The aerodynamic test was performed in the fixture shown in Figure 24a. The door was supported at the two hinges and latches, and a uniform pressure was applied by means of water bags and sand bags. In addition, loads from the cabin door were applied at the cabin-to-litter door hinges.

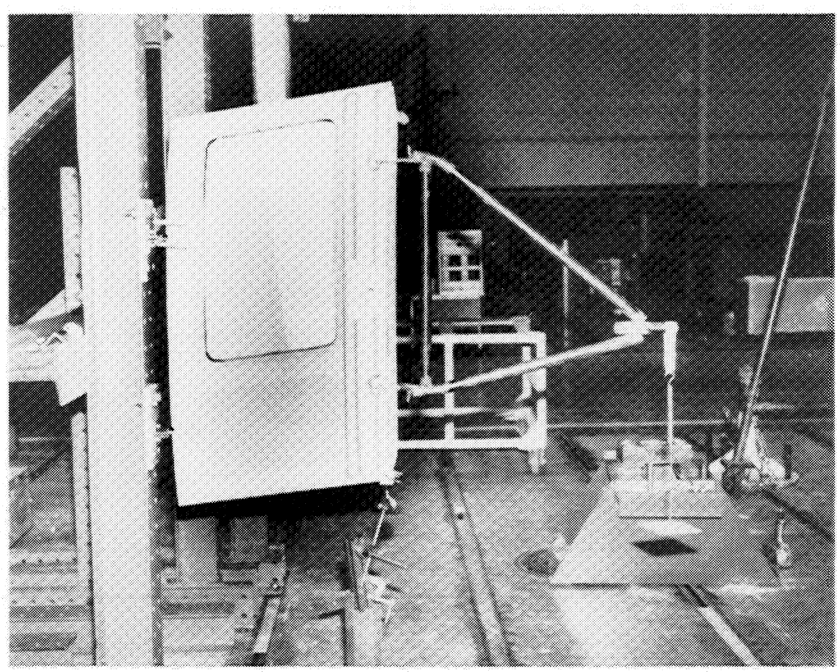
Failure occurred at 0.55 psi pressure (634 pound), 152 pounds at the upper hinge and 390 pounds at the lower hinge. This is between 95 and 99 percent of the required strength if the failure were in the composite material. However, failure was caused by the hinge pin slipping out of the latch, not by failure of the composite material, and for this failure the knockdown factor is not applicable. On this basis the strength of the door was certified by the FAA.

The cantilever test setup is shown in Figure 24b. At 415 pounds of load the door hinges rotated excessively to the point where it was not possible to load any higher in this manner. Since a failing load test was required, the load application point was moved directly to the litter door hinges. This reduces the cantilever moment, but eliminates the rotation of the hinges. Failure occurred at 833 pounds when loaded in this manner, and the failure occurred by compression buckling at the lower forward corner.

The deflections of two points on the fore and aft centerline of the litter door are shown in Figure 25 for the aerodynamic loading condition. The points shown on the figure are the average of two tests, and are the sum of the distributed load and the two concentrated hinge loads. The data is linear to limit load, which is the maximum load at which deflections were measured. The deflection of point 2 is higher than for point 1 because of its close proximity and the concentrated hinge loads.



a. Pressure test



b. Cantilever test

Figure 24. Static tests of composite litter door.

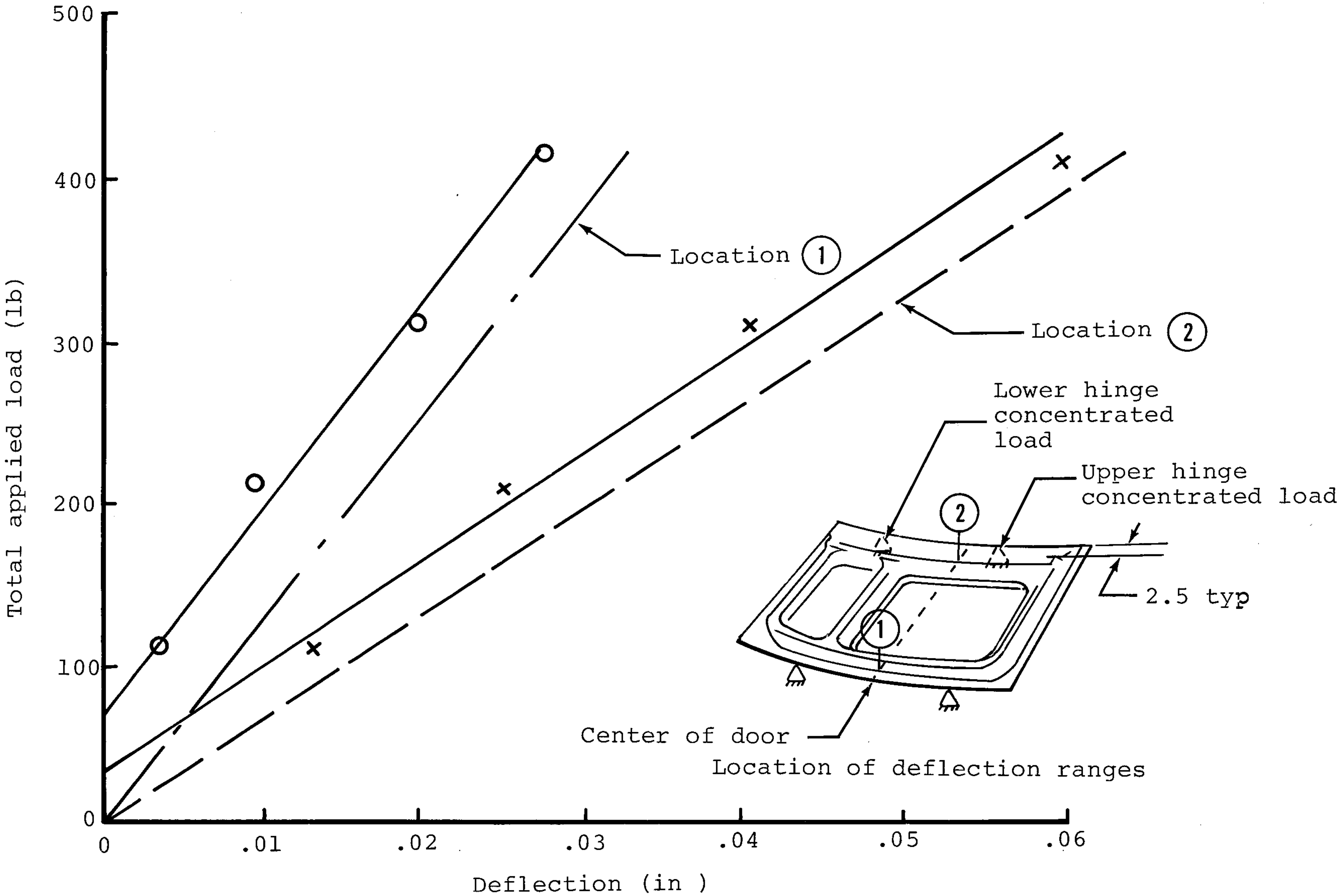


Figure 25. Load deflection curves of two points on the litter door.

6.2.3 Forward Fairing Loads and Tests

A single test condition, that of aerodynamic pressure, was established for the fairing:

Aerodynamic Limit Load	=	0.20 psi acting outboard
Ultimate Load	=	0.30 psi
Required Strength	=	Ult. x knockdown factor (1.62)
	=	0.49 psi

Figure 26 shows the test setup for the fairing. The fairing was attached to the test fixture at the forward hinge points and aft latches. The cover was closed and sealed. Air was then evacuated from the jig, causing a vacuum on the outer surface of the fairing. Air was evacuated until failure occurred.

Two fairings were tested to failure. The first one failed at 1.51 psi at the inner corner splice, as shown in Figure 27. The second fairing failed at 1.94 psi, but was not a composite failure. At 1.94 psi the right-hand aft latch slipped, causing some local, noncatastrophic failures of the surrounding structure.

Subsequent examination of the test parts revealed a possible knife cut in the failed area of the first specimen and the probability of a faulty latch on the second specimen. (The much higher load that five subsequent fairings withstood supports this theory. This is discussed in Section 6.2.6).

Since the failing pressures of 1.51 and 1.94 psi were considerably higher than the required strength of 0.49 psi, the fairing was certified by the FAA.

Figure 28 shows the load-deflection curve of the top centerline of the fairing to limit load for the first test.

6.2.3.1 Effects of Low Curing Temperature on the Forward Fairing. In Section 5.1 it was stated that the cure of the forward fairing faces and the bond to the Klegecell foam was accomplished at 200°F. The standard cure temperature for the Kevlar/epoxy used for the forward fairing is 260°F. Because of the lower cure temperature, there was concern that the Kevlar/epoxy might have a tendency to absorb excessive amounts of moisture which would result in a decrease in strength.

As a screening test, specimens of Kevlar/epoxy fabric that had the 200°F 5-hour cure were evaluated against those that had the 260°F 1.5-hour cure by immersing them in water and comparing moisture absorption of each as a function of time. Figure 29 shows that, instead of absorbing more moisture, the 200°F cured specimens actually absorbed less. A possible explanation may be

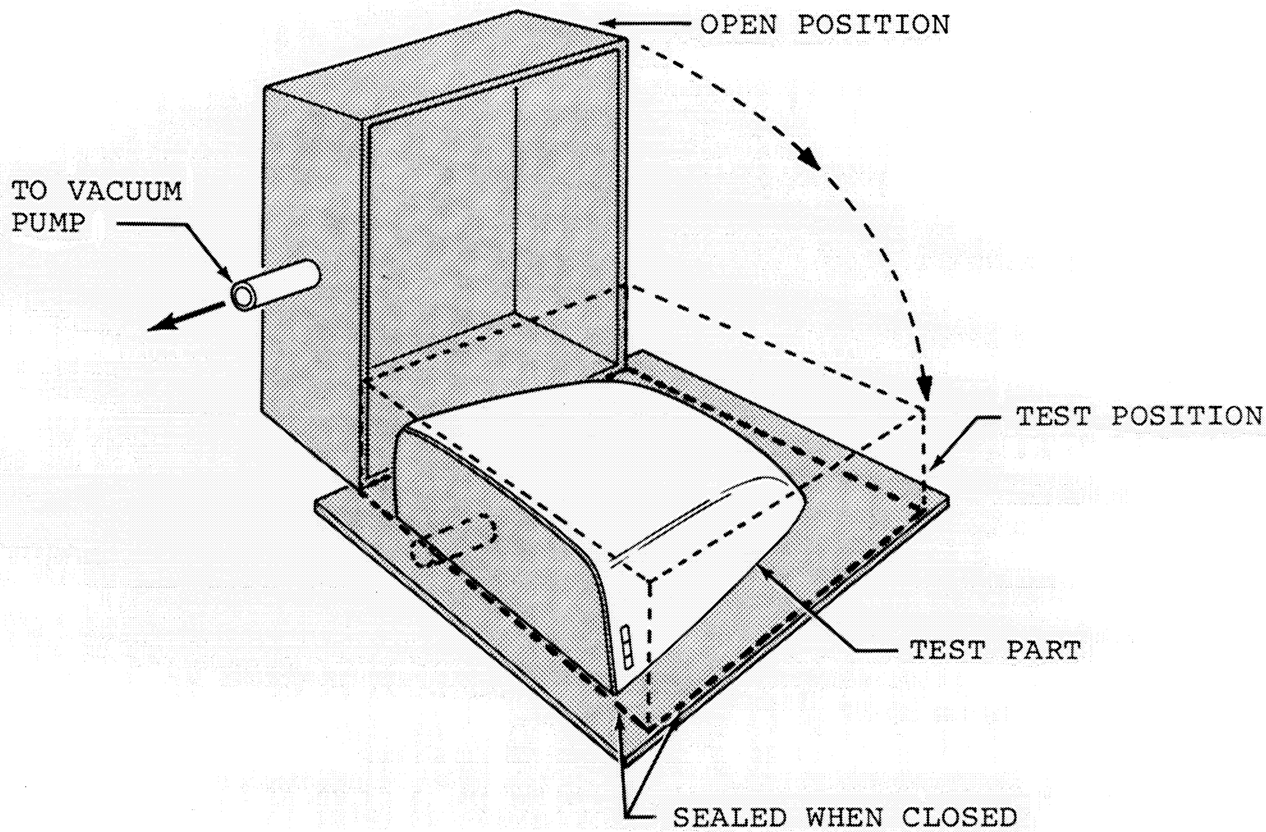


Figure 26. Pressure test setup on forward fairing.

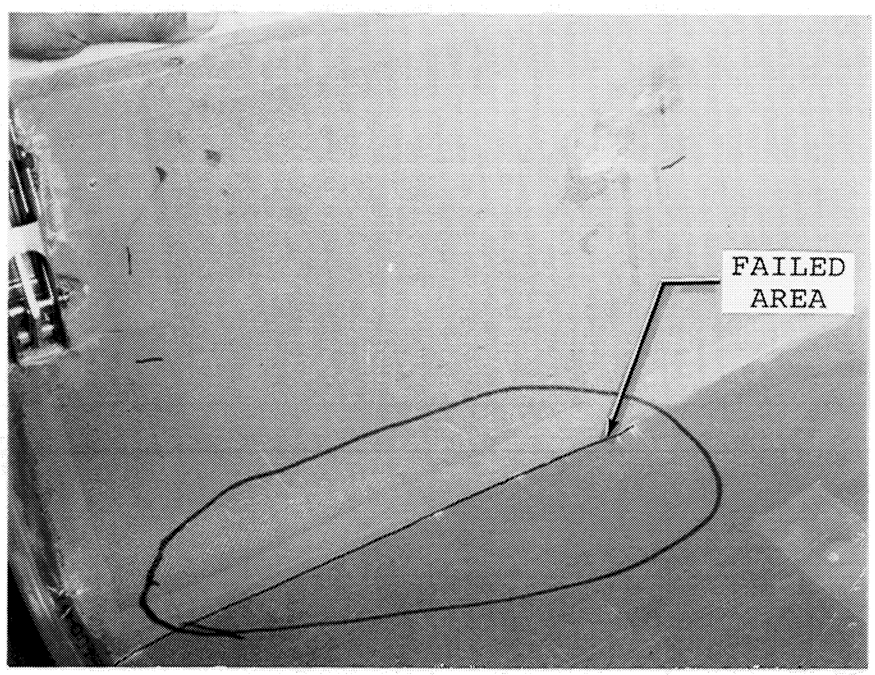
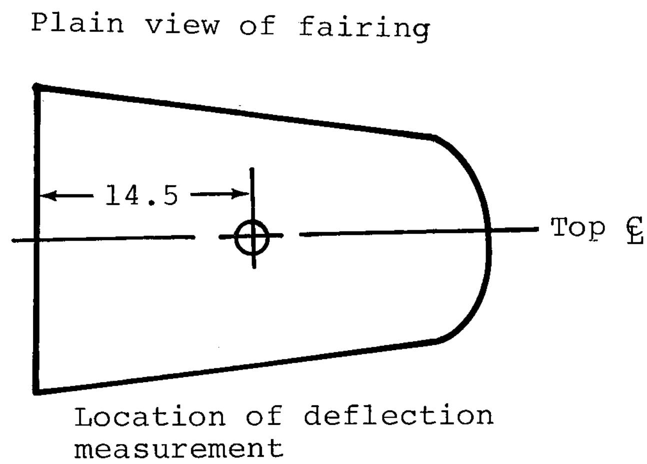


Figure 27. Forward fairing failure.



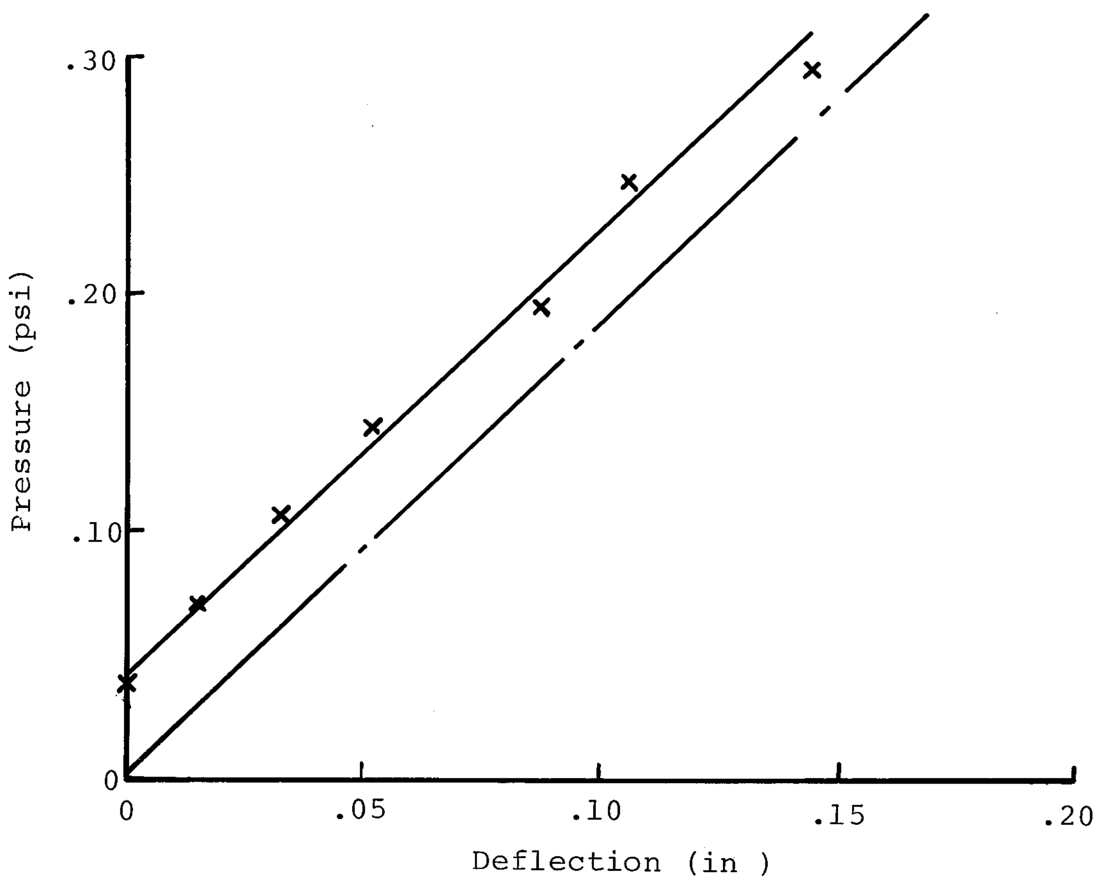


Figure 28. Deflection of top centerline of forward fairing during pressure test.

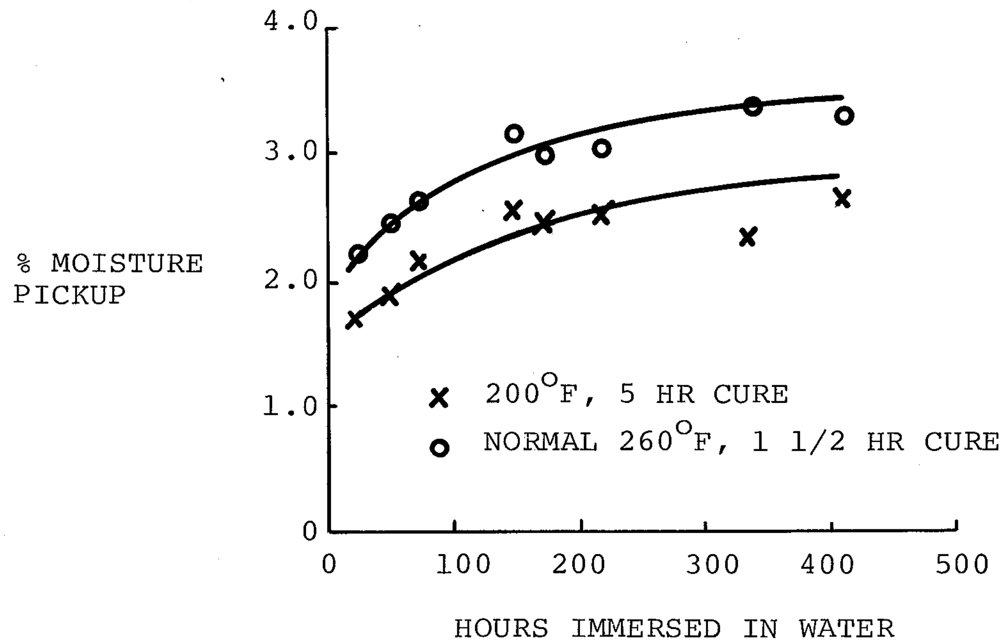


Figure 29. Test for moisture pickup. Kevlar/epoxy fabric immersed in 120°F water.

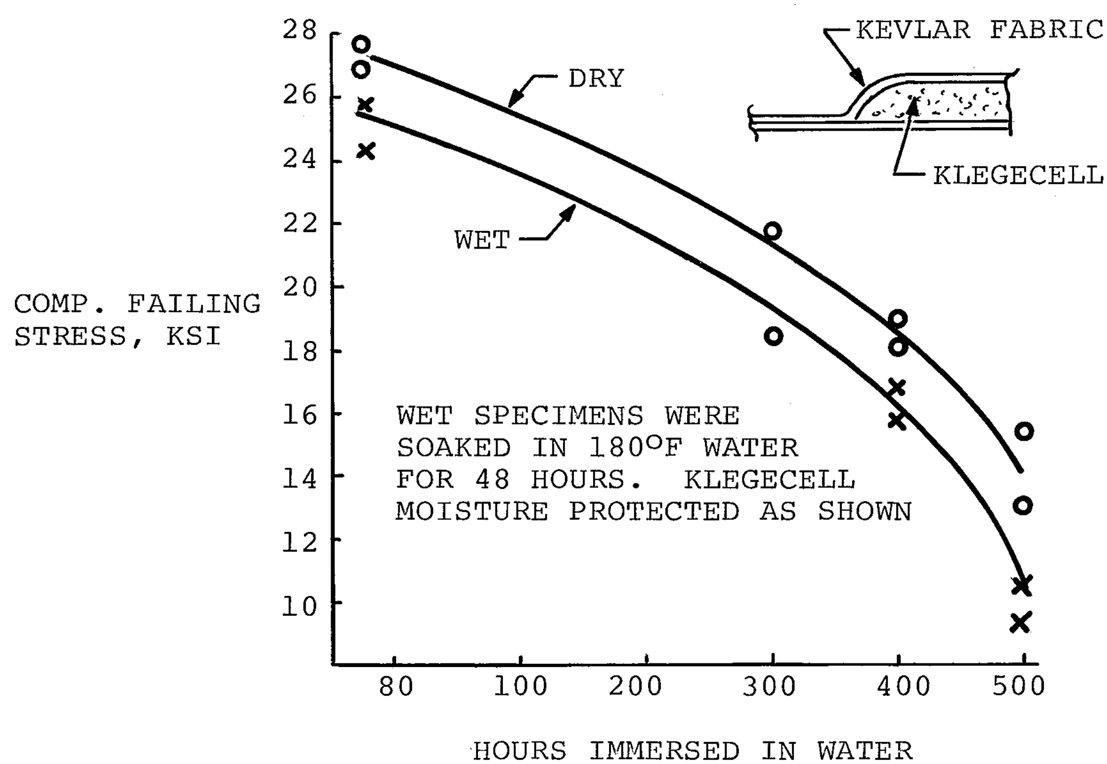


Figure 30. Comparison of wet and dry bending strength of simulated forward fairing panels.

that the 200°F system was cured for 5 hours as compared with 1.5 hours for the normal 260°F cure. The longer cure time could result in a more moisture-resistant resin.

Another series of tests were performed to compare the effects of temperature on wet and dry specimens of the sandwich structure. Figure 30 shows the results of 4-point loading tests made on beams representative of the fairing structure. All of the specimens were made from the same panel. For purposes of these comparative tests, the wet specimens were soaked in 180°F water for 48 hours. All failures were in the compression facesheet. The differences between wet and dry specimens were about as expected, and were well within design requirements. Accordingly, it was concluded that the lower cure temperature does not affect the moisture pickup or elevated temperature problem to any significant extent.

6.2.4 Vertical Fin Test Program

6.2.4.1 Fin Static Loads. The certification tests for the graphite/epoxy vertical fin consisted of two failing load tests and a fatigue test program. The static failing load tests were conducted on full fins, and the fatigue tests were conducted on test specimens that simulated the fin-to-fuselage attach structure.

At the beginning of the program it was planned to perform all tests at 180°F and ambient humidity as soon after being removed from environmental conditioning as possible. (Environmental conditioning was defined as 42 days soak at 120° ± 5°F and 95 ± 5 percent relative humidity.) It was necessary to alter this plan to test at ambient conditions. The reasons for this change are discussed in Par. 6.2.4.3.

The following two conditions were specified for static failing load tests:

a. Aerodynamic Loading

Aerodynamic Limit Pressure (uniformly distributed over the fin)	=	0.50 psi
Ultimate Pressure	=	0.75 psi
Fin Area	=	1387 in^2
Limit Aerodynamic Load	=	693.5 lb
Ultimate Load	=	1040 lb

b. <u>Tail Down Landing Load Applied at Aft End of Tail Skid</u>

Limit Vert Tail Skid Load	=	273 lb
Limit Lateral Load	=	136.5 lb
Resultant Limit Load	=	305.2 lb
Reserve Energy Vert Tail Skid Load	=	315 lb
Reserve Energy Lateral Load	=	157.5 lb
Resultant Reserve Energy Load	=	352.2 lb

6.2.4.2 <u>Fin Fatigue Loads</u>. The fatigue test was performed at room temperature ambient conditions as soon as possible after being environmentally conditioned. The loads were 125 percent of the maximum load found during the Model 206L flight loads survey for level flight high speed. The setup is shown schematically in Figure 31. Four test specimens were required, and 10 million cycles of load were required for each specimen without failure.

6.2.4.3 <u>Fin Static Test Results</u>. In the course of environmentally conditioning the fin, a portion of the structure got too close to a heating element. Discoloration of part of the lower area of the fin indicated that some of the structure was overheated. This particular fin was used for the tail skid test. At 240 pounds, or 78.6 percent of the 305.2-pound limit load, the local area where the tail skid attaches to the skin failed.

Examination of the failed area revealed two facts: first, that the area had been overheated and was probably partially disbonded before the test was started. Second, that the overheated area was sufficiently localized for the fin to be used for the aerodynamic test.

In the interest of performing the aerodynamic test as soon as possible so as not to lose the environmental conditioning, the FAA agreed to testing at room temperature and to the use of a knockdown factor based on the BHTI-developed data for T300/788 shown in Table 12.

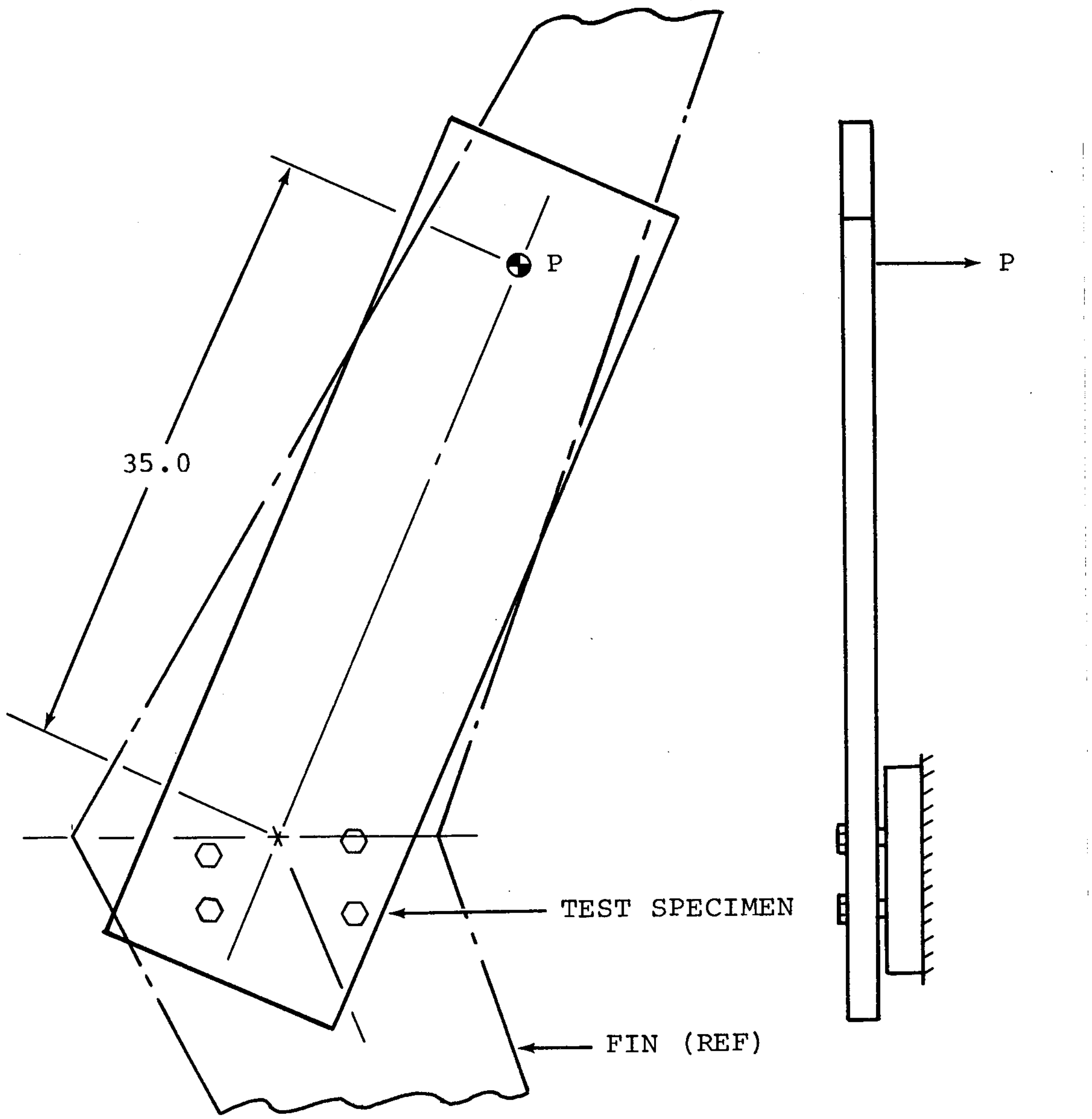


Figure 31. Setup for fin fatigue test.

TABLE 12. BHTI TEST DATA FOR T300/788 GRAPHITE/EPOXY

Property	Average Room Temp Dry (ksi)	Average 180°F Wet (ksi)	Knockdown Factor
0° Tension	255.4	251	1.02
0° Compression	137.6	108.2	1.27
In-Plane 0°-90° Shear	10.7	7.98	1.34

An apparent knockdown factor of 1.34 could be used, but a factor of 1.40 was agreed upon with the FAA.

The fin was loaded uniformly with shot bags. Failure occurred approximately 10 inches above the upper set of fuselage attach bolts in bending compression of the facesheet, as shown in Figure 32. The failing load was 2025 pounds. The required strength was 1040 x 1.4 = 1456 pounds, which gives a margin of safety of 39 percent for this condition.

Since an apparent knockdown factor was already established, it was agreed to retest the tail skid loading condition at room temperature ambient conditions. Failure occurred at a resultant load of 927 at the tail skid by buckling of the compression skin, as shown in Figure 33.

$$\text{Required strength} = 1.4 \times 352.2 = 493 \text{ lb}$$

The margin of safety is 88 percent for this condition.

6.2.4.4 Fin Fatigue Test Results. Four specimens were tested in a Sonntag fatigue testing machine at a frequency of 1800 cycles per minute. The test configuration simulated the shears and bending moments at the fin-to-fuselage attachment. Only the shears and moments from the upper fin were applied. The loads from the lower fin were conservatively neglected since they would subtract from the attachment loads. The structure at the attachments was an exact simulation of the fin-to-fuselage structure. Figure 34 shows the test specimen.

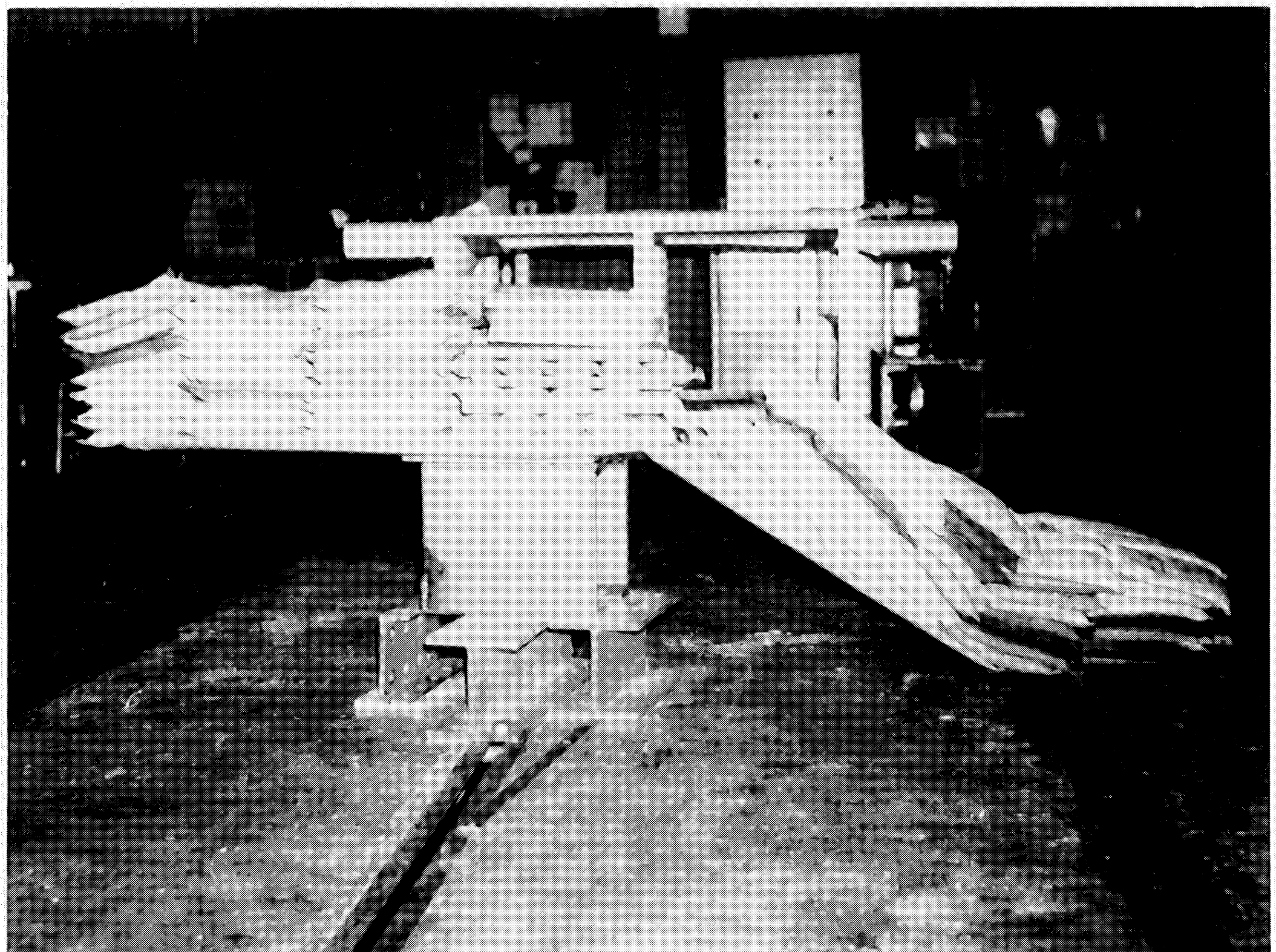


Figure 32. Fin static test failure - aerodynamic loading.

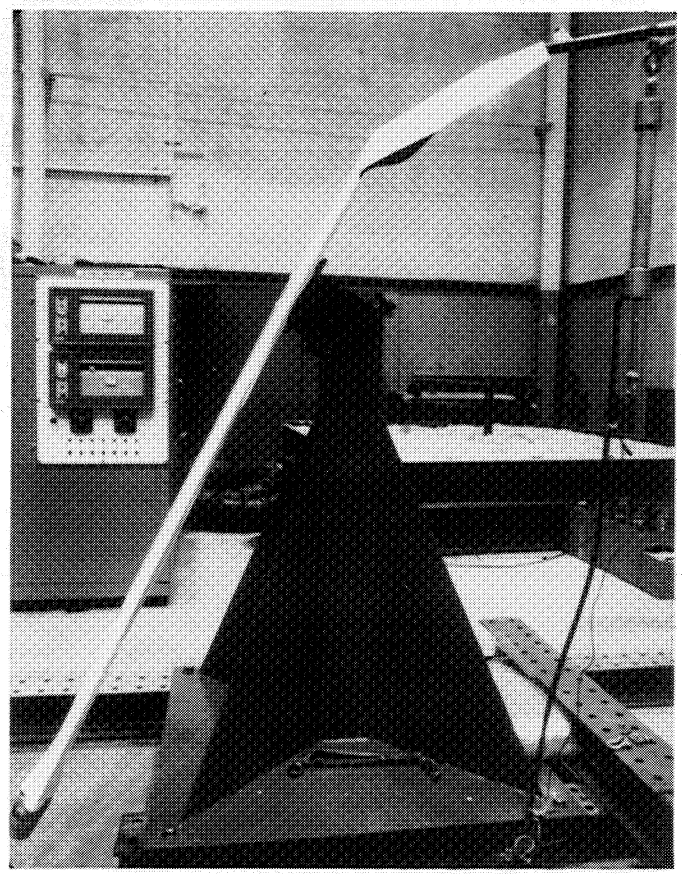


Figure 33. Fin static test failure - tail down landing.

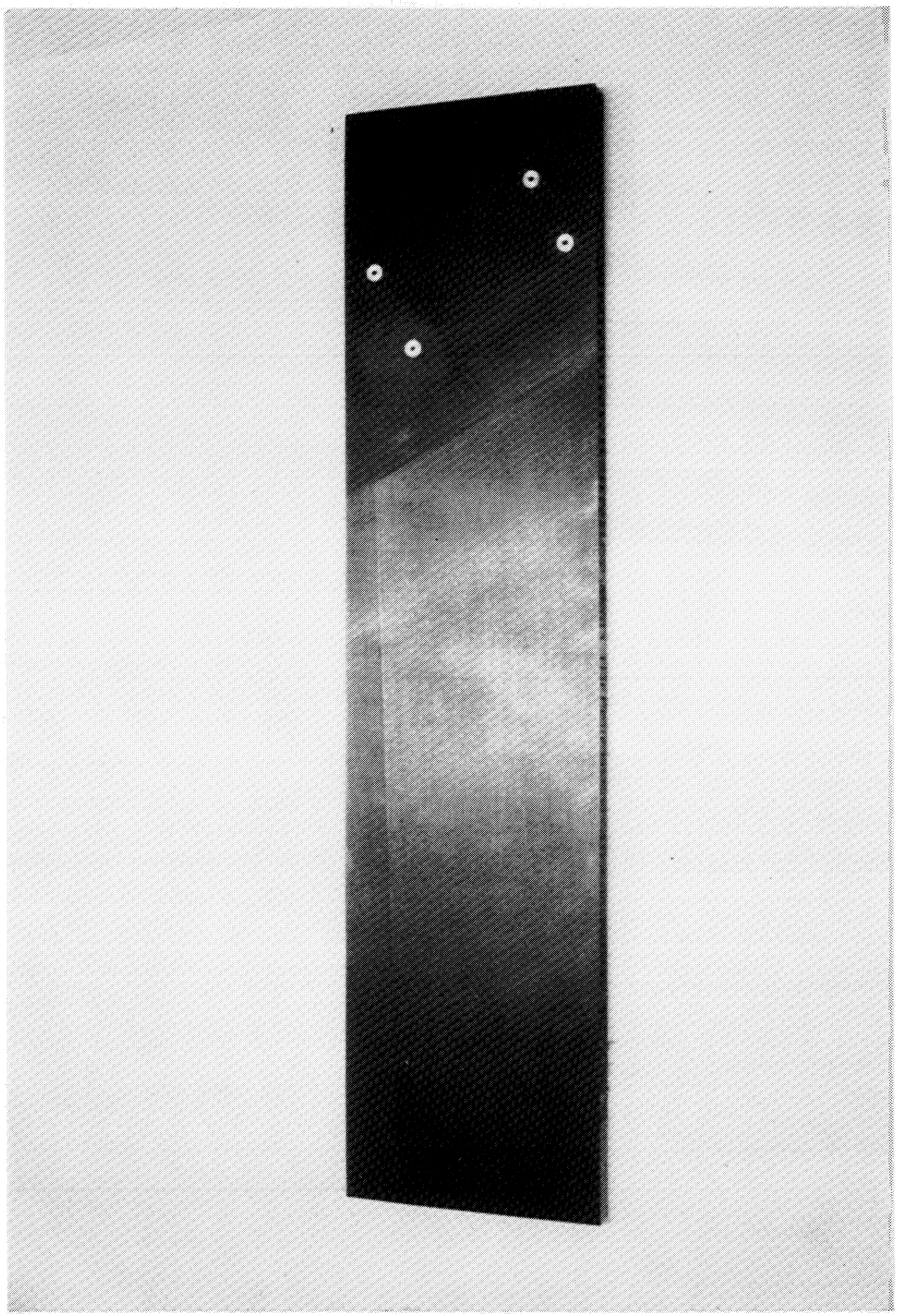


Figure 34. Fin fatigue test specimen.

The results of the fatigue test program are shown in Table 13. Although only 10 million cycles of load were required for each specimen, the first three specimens were cycled an additional 5 to 7 million cycles with no failure.

On the final specimen the oscillatory load was increased by 50 percent after the required 10 million cycles was reached, in an attempt to obtain a failure. After an additional 35 million cycles, there was no indication of failure so the test was terminated.

TABLE 13. FATIGUE TEST RESULTS FOR COMPOSITE VERTICAL FIN

Specimen No.	Moment (in-lb)	Test Cycles
1	1960 ± 1610	17,551,000
2	1960 ± 1610	15,142,000
3	1960 ± 1610	14,979,000
4	1960 ± 1610 1960 ± 2430	10,398,000 34,758,000

6.2.5 Static Tests of Production Components

As part of the program to evaluate the quality of production parts, five of each component were randomly selected from the completed stock and tested to failure. The components received no environmental preconditioning, and the tests were performed at laboratory ambient conditions because these tests were a check of manufacturing quality. It was only necessary that the five components be tested under identical conditions. In all cases the tests were performed on the same test fixtures and for the same aerodynamic pressure conditions that were used for the FAA certification tests.

The results of the random component tests are shown in Table 14. The table shows that only four baggage doors were tested to failure. The fifth test had been stopped prematurely when a failure was erroneously reported. When the part was cut up to examine what was thought to be a failure, the area was found to be undamaged. This was verified by a subsequent local element test. However, since there were no spare doors available for additional tests, the strengths of only four doors could be reported.

Table 14 shows that the average failing pressure for the production forward fairings was about 81% higher than for the certification fairings. It was noted in Section 6.2.3 that the failing pressures for the two certification fairings were lower than predicted, and that there appeared to be a knife cut in one specimen, and a defective latch on the other. This is borne out by the higher strengths exhibited by the random production fairings.

TABLE 14. FAILING LOAD TESTS OF RANDOM COMPONENTS

Component		Failing Load (lb)	Certification Failing Load (lb)
Baggage Door	#1	550	
	2	650	
	3	700	
	4	551	
Average		613	695
Litter Door*	#1	1347	
	2	1237	
	3	1170	
	4	1070	
	5	1250	
Average		1215	1176
Forward Fairing**	#1	3.06	
	2	3.00	
	3	3.40	
	4	2.20	
	5	3.00	
Average		3.13	1.73
Vertical Fin	#1	2025	
	2	1872	
	3	1900	
	4	2122	
	5	2177	
Average		2097	2025

* Sum of aerodynamic pressure and concentrated loads at passenger door hinges
** Pressure in psi.

7. CONCLUDING REMARKS

1. Forty-five shipsets of primary and secondary advanced composite structures were built in standard helicopter production environments using normal production practices. The quality of the parts was consistent with only small variations in strength (except for the forward fairing) between certification parts and randomly chosen production parts.

2. For the forward fairing, litter door, and vertical fin, the fabrication learning curve did not show an appreciable reduction in labor hours. A significant reduction in fabrication man-hours was acheived for the baggage doors when fabricated by the same personnel. Increases in fabrication man-hours were evident when new personnel were assigned to the job.

3. The weights for fill-and-fair and paint were higher than expected. For thin-gage structure, these weights tend to offset a significant portion of weight savings achievable through the use of advanced composites.

4. Warpage problems were encountered during fabrication of the litter doors. The doors were straightened through judicious application of heat and pressure.

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1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR-166002		

4. Title and Subtitle	5. Report Date
Flight Service Evaluation of Composite Components on the Bell Helicopter Model 206L: Design, Fabrication, and Testing	November 1982
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15. Supplementary Notes

Langley Technical Monitor: D. J. Baker
Technical Summary Report

16. Abstract

This is an interim report on a program to obtain long term flight service experience on four 206L helicopter advanced composite airframe components. Forty-five ship sets of components were built. The report discusses the work performed prior to first flight of the components. This includes design, fabrication, and FAA certification. It also includes a program to track the cost of the parts. In addition, five exposure racks containing a total of approximately 2000 specimen were built and located at selected sites in the United States and Canada. This program is also discussed.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Flight service, advanced components, FAA certification, composite specimen environmental exposure	Unclassified - Unlimited Subject Category 24

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