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Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft

DRL 003

QUARTERLY TECHNICAL REPORT – NO. 12

This report is for the period 1 October 1978 through 31 December 1978

Lockheed Corporation
Lockheed-California Company
Post Office Box 551
Burbank, California 91520

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Prepared for Langley Research Center

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Approved By: Fred English
F. C. English
Program Manager

22 January 1979

Prepared for Langley Research Center

FOREWORD

This report was prepared by the Lockheed-California Company, Lockheed Corporation, Burbank, California, under contract NAS1-14000. It is the 12th quarterly technical report, covering work completed between 1 October 1978 and 31 December 1978. The program is sponsored by the National Aeronautics and Space Administration (NASA), Langley Research Center. The program manager for Lockheed is Mr. Fred C. English. Mr. Louis F. Vosteen is project manager for NASA, Langley. The technical representative for NASA, Langley is Mr. Herman L. Bohon.

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SUMMARY

The technical activities performed in this reporting quarter and documented in this report are related to tasks associated with Phase II, Phase III, and Phase IV of the Advanced Composite Vertical Fin (ACVF) Program. These tasks include the following: in Phase II, Component Definition, Material Verification, Process Verification, and Concept Verification; in Phase III, Cover and Spar Fabrication and Test Support; and in Phase IV, Component Tool Development.

Work on process verification and tooling development continued in this reporting period. The cover process development was completed with the decision to proceed with low resin content prepreg material (34 ±3% by weight) in the fabrication of production readiness verification test (PRVT) specimens and the full-scale covers.

The structural integrity of the cover/joint design was verified with the successful test of the cover attachment to fuselage ancillary test specimen (H25). Failure occurred, as predicted, in the skin panel away from the fuselage joint at 141 percent of the design ultimate load.

With the successful completion of the H25 test, the PRVT cover specimens, which are identical to the H25 ancillary test specimen, were cleared for production. Eight of the twenty cover specimens have been fabricated and are in preparation for test. All twenty of the PRVT spar specimens have been fabricated and are also being prepared for test. The environmental chambers to be used in the durability test of ten cover and ten spar PRVT specimens have been completed and installed in the load reaction frames. Facility checkout with spars installed in the environmental chambers will be initiated during the next reporting period.

The first full-scale front spar made from graphite/epoxy was fabricated successfully. The tools for the full-scale rear spar are nearing completion. The full-scale cover tools are almost complete and are undergoing thermal and

vacuum checks. Fabrication of both the rear spar and the covers will commence in the next reporting period.

The indicated weight saving for the ACVF is currently at 27.9 percent (239.1 pounds) including a 10-pound growth allowance. Without the growth allowance, a weight saving of 29.0 percent (249.1 pounds) is anticipated. Composite material utilization is currently predicted to be 77.8 percent of the redesigned fin box weight.

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SECTION 1

INTRODUCTION

The broad objective of the Aircraft Energy Efficiency (ACEE) Composite Structures Program is to accelerate the use of composite structures in new aircraft by developing technology and processes for early progressive introduction of composite structures into production commercial transport aircraft. The program, as one of several which are collectively aimed toward accomplishing that objective, has a specific objective: to develop and manufacture advanced composite vertical fins for L-1011 transport aircraft. Laboratory tests and analyses will be made to substantiate that the composite fin can be safely and economically operated under service loads and environments and will meet FAA requirements for installation on commercial aircraft. A limited quantity of units will be fabricated to establish manufacturing methods and costs. The Advanced Composite Vertical Fin (ACVF) will make use of advanced composite materials to the maximum extent practical and weigh at least 20 percent less than the metal fin it replaces. A method will be developed to establish cost/weight relationships for the elements of the composite and metal fins to establish cost effective limits for composite applications.

The ACVF to be developed under this program will consist of the entire main box structure of the vertical stabilizer for the L-1011 transport aircraft. The box structure extends from the fuselage production joint to the tip rib and includes the front and rear spars; it is 25 feet tall with a root box chord of 9 feet and represents an area of 150 square feet.

The primary emphasis of this program is to gain a high level of confidence in the structural integrity and durability of advanced composite primary structures. An important secondary objective is to gain sufficient knowledge and experience in manufacturing aircraft structures of advanced composite materials to assess properly its cost-effectiveness.

The duration of this program is 70 months, with completion scheduled November 1982. The master schedule for this program is shown in Figure 1-1. The program is organized in four overlapping phases: Phase II - Design and Analysis; Phase III - Production Readiness Verification Tests (PRVT); Phase IV - Manufacturing Development; and Phase V - Ground Tests and Flight Checkout. Phase I was completed during 1976.

The Lockheed-California Company has teamed with the Lockheed-Georgia Company in the development of the ACVF. Lockheed-California Company, as prime contractor, has overall program responsibility and will design and fabricate the covers and the ribs, conduct the PRVT program, and conduct the full-scale ground tests; Lockheed-Georgia Company will design and fabricate the front, rear, and auxiliary spars, and assemble the composite fin at the plant in Meridian, Mississippi, where the present L-1011 vertical fins are assembled.

Phase I, Engineering Development, has been completed and Phases II, III and IV are in progress.

Phase II, Design and Analysis, consists of completing the detail design and analysis, characterization of the T300/5208 material system, initiating producibility studies, and conducting material, process, and concept verification tests. Phase III, Production Readiness Verification Testing (PRVT) is designed to provide information to answer the following questions:

- What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality levels and quality control procedures?
- What variability in static strength can be expected for production quality components, and are the margins sufficient to account for this variability?
- Will production quality components survive extended-time laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence in in-service durability?

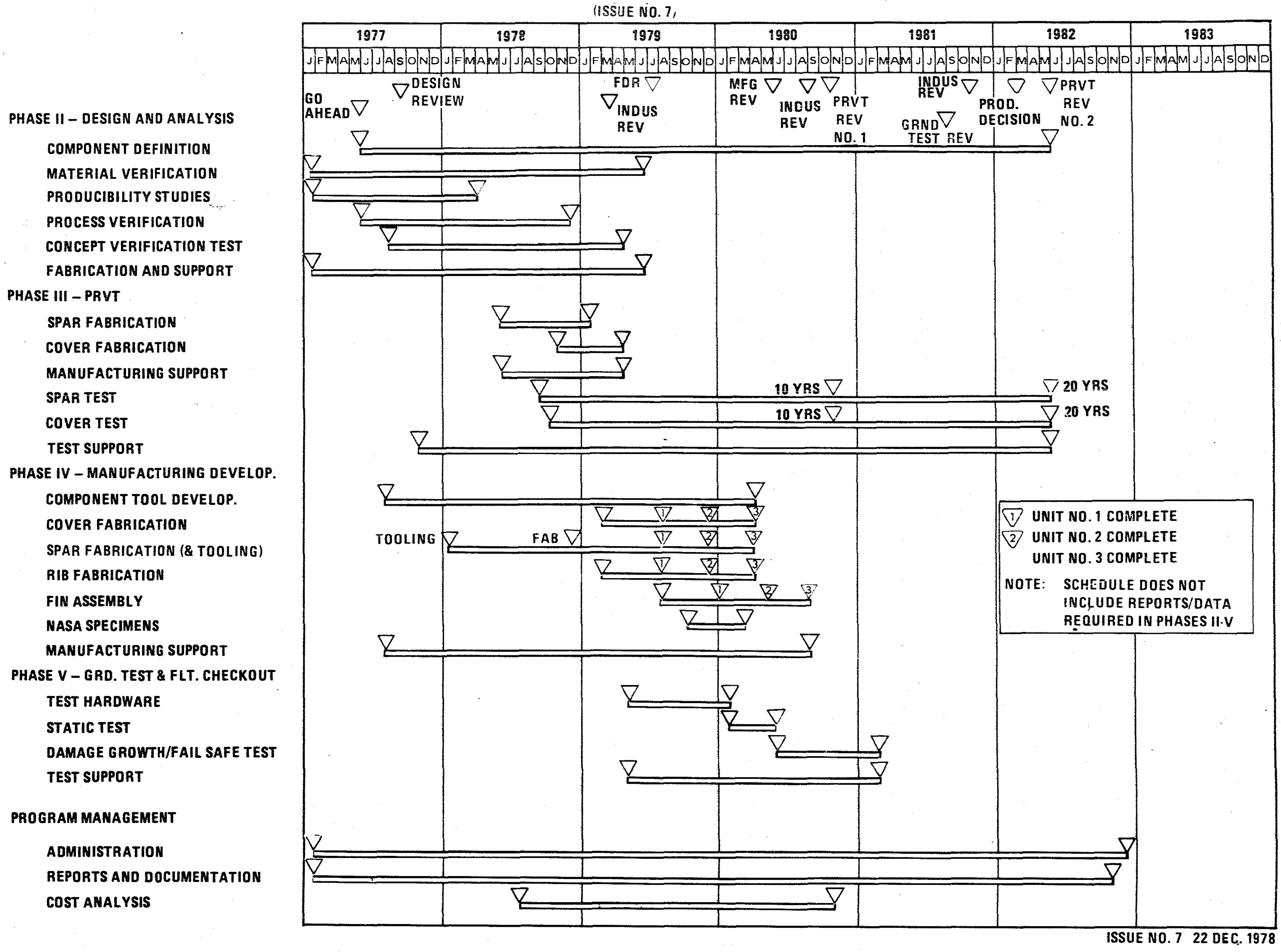


Figure 1-1. ACVF Program Master Schedule

Ten static strength tests and ten durability tests will be conducted on each of two key structural elements of the ACVF. One element will represent the front spar/fuselage attachment area, and the other element will represent the cover/fuselage joint area.

Manufacturing Development, Phase IV, conducted concurrent with Producibility Studies and Process Verification Tasks, will accommodate changes in tooling to take advantage of development of low-cost manufacturing methods. Following NASA's approval of the design, three fins will be fabricated and assembled to prove the design, methods of manufacture, and quality. Actual costs will be documented during fabrication and components will be weighed to update cost and weight estimates.

The manufacturing cost history obtained through the fabrication of the PRVT specimens in a production environment will provide cost data for a starting point for this application of composite structure. Together, they will form the basis for reasonably confident estimates of future production costs.

Ground tests will be conducted on two full-scale fin box beam structures mounted on simulated fuselage support structures during Phase V. The test plan will include static tests, ultimate load and failure load tests on one GTA. Damage growth tests to two lifetimes, and fail-safe and residual strength tests will be done on the second GTA. Repair techniques for in-service maintenance and inspection will be employed throughout tests. Test results will be used to verify the analytical, design, and fabrication procedures, and are essential inputs to the FAA for certification of the aircraft with the ACVF installed. Ceritification will be based on satisfying both static strength and fail-safe requirements.

Throughout this program, technical information gathered during performance of the contract will be disseminated throughout the aircraft industry and Government. The methods used to distribute this information will be through Quarterly Reports, which will coincide with calendar quarters; and Final Reports of each phase to be distributed at the completion of each phase. All test data and fabrication data will be recorded on Air Force Data Sheets for incorporation in the Air Force Design Guide and Fabrication Guide for Advanced Composites. Oral Reviews will also be conducted at NASA, Langley to acquaint the aircraft industry and the Government with progress of the program.

SECTION 2

PHASE II - DESIGN AND ANALYSIS

Phase II, design and analysis, comprises the main engineering effort in the design and manufacturing development of the spars, ribs and cover assemblies for the L-1011 composite vertical fin. The effort during this reporting period covered five tasks: component definition, material verification, process verification, concept verification, and quality assurance.

2.1 COMPONENT DEFINITION

Component definition covers the detail design and structural analysis of the spar, rib and cover configurations. Detail design has been completed.

2.1.1 Weight Status

The current weight status is shown in Table 2-1 and is unchanged from the last quarter. A weight savings of 27.9 percent (239.1 pounds) is currently being predicted including a 10 pound growth allowance. Without growth allowance a weight savings of 29 percent (249.1 pounds) is anticipated. Composite material use is currently predicted to be 77.3 percent of the redesigned fin box weight. A weight-time history for the composite fin is provided in Figure 2-1.

2.2 MATERIAL VERIFICATION

This task is structured to develop the material properties of the T300/5208 unidirectional tape material system to derive design allowables for the ACVF.

TABLE 2-1. CURRENT WEIGHT STATUS

Item	Metal Design Total Weight (lb)	Composite Design			Weight Change
		Target Weight (lb)	Total Weight (lb)	Composite Mat'l Wt (lb)	
Covers	460.4	368.4	351.7	333.9	-2.0
Spars	199.0	132.0	117.2	87.9	-0.3
Ribs	153.3	131.8	107.0	46.1	-1.0
Assembly Hardware	35.4	16.7	14.6	-	+0.2
Protective Finish	9.6	9.6	9.6	-	
Lightning Protection	-	15.5	0.0	-	-14.2
Installation Penalty	-	5.0	8.5	-	
Design Growth Allowance	-	-	10.0	10.0	-14.0
Total Fin Predicted Delivery Weight - lb	857.7		618.6	477.9	-31.3
Weight Saving - lb			239.1		
Percent Weight Saved			27.9		
Percent Composite Material				77.3	
Total Fin Current Ind Indicated Weight - lb (Predicted Less Growth)		679.0	608.6 29.0	467.9 76.9%	
Current Indicated Weight of Redesigned Component	8.25.4 [1]		587.4	(28.8% Weight Saved)	[2]

Weight Basis: 5% EST, 95% CALC, 0% ACT

[1] Total metal design weight less weight of components not redesigned

[2] Based on redesigned metal components

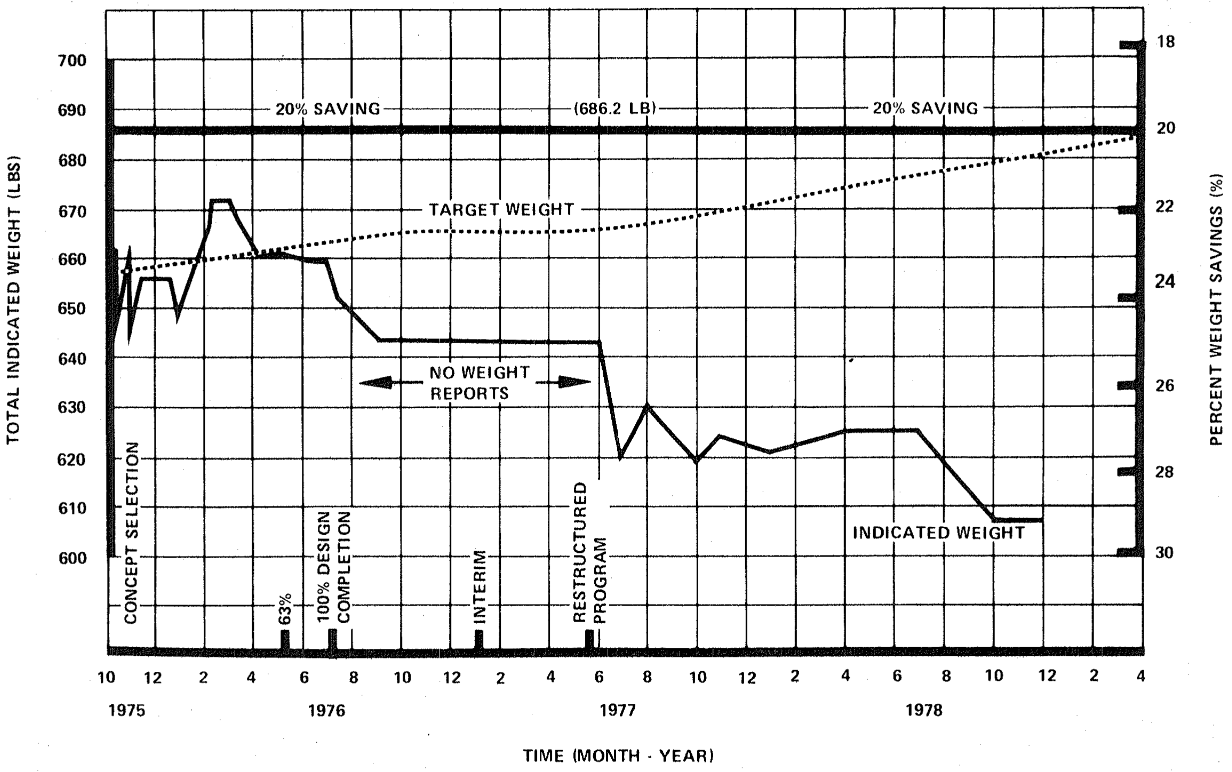


Figure 2-1. Weight-Time History

2.2.1 Evaluation of Defect Tolerance in Composites - Test Item H12B

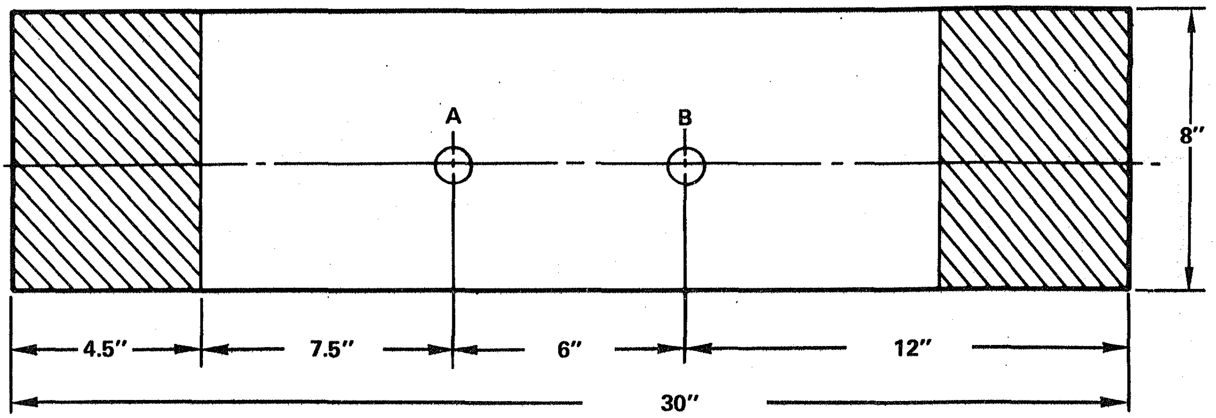
Evaluation of defect tolerance in composites, Item H12B had the objective of assessing the tolerance to defects in T500/5208 composite laminates.

This test program has been completed. Eight specimens of the configuration shown in Figure 2-2 have successfully completed four lifetimes of fatigue loading with no propagation of the imbedded defects. Four specimens were tested at room temperature in ambient air, and four specimens, preconditioned to 1% moisture gain, were tested under the environmental cycle shown in Figure 2-3.

2.2.2 Graphite Epoxy Laminate Durability - Test Item H13D

The objective of this test item is to determine the durability of laminates, typical of L-1011 fin application, when fatigue tested under the environmental cycle shown in Figure 2-3. Static tests of unnotched and notched

16-PLY LAMINATE $(\pm45/0/\mp45/\pm45/0)_S$ T300/5208



DEFECT: 1 INCH DIA. 0.0005 INCH KAPTON

DEFECT A LOCATED BETWEEN 2ND AND 3RD PLYS

DEFECT B LOCATED BETWEEN 8TH AND 9TH PLYS

CROSS-HATCHED REGION: FIBERGLASS END TABS

Figure 2-2. Defect Tolerance System

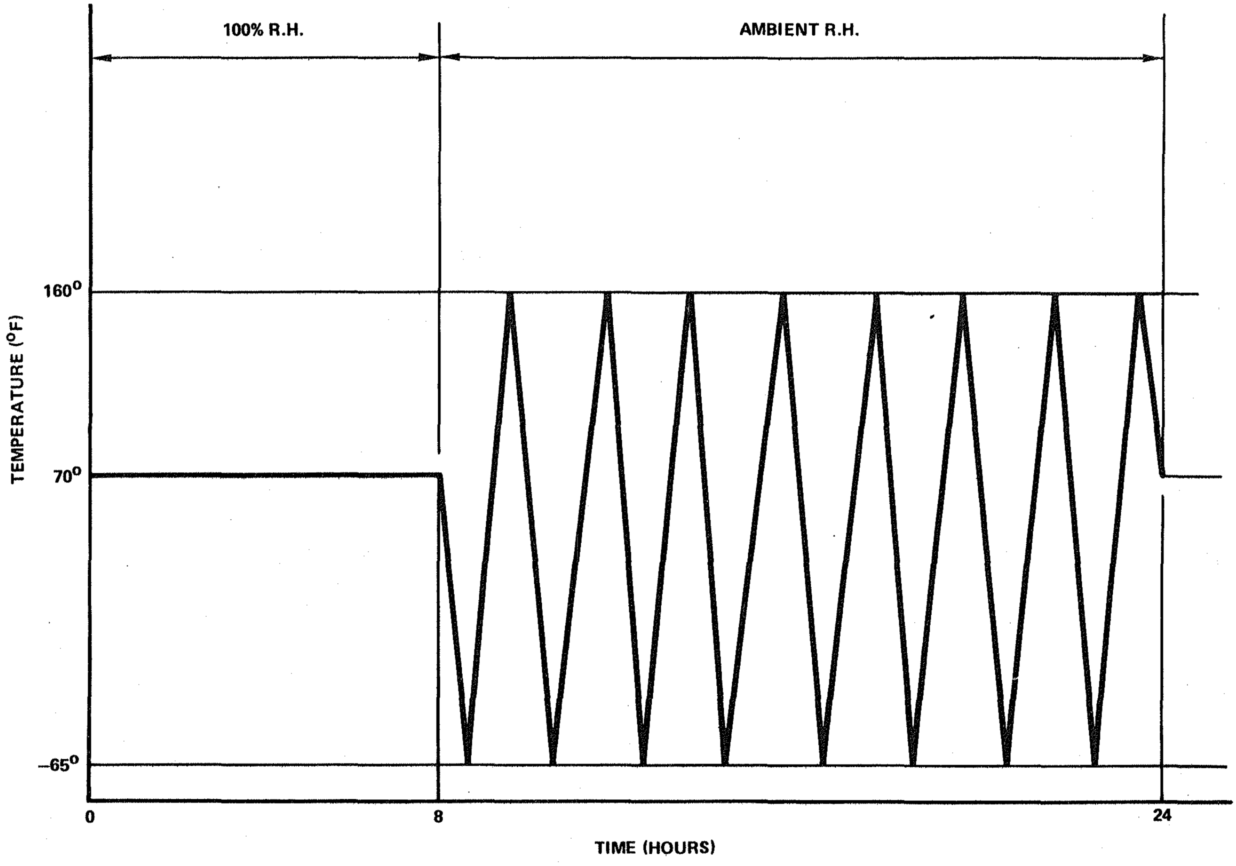


Figure 2-3. Typical Environmental Cycle

coupons will be conducted to compare with the results of the residual strength tests conducted on the durability coupons. The number of specimens to be tested is shown in Table 2-2. All test coupons have been fabricated and are being prepared for test. Testing of the static and durability coupons will start early next year.

2.3 PROCESS VERIFICATION

Process verification activity on the covers was concluded in this reporting period with the fabrication of a series of hat-stiffened panels utilizing low resin and high resin content prepreg material. A producibility evaluation of these panels led to the selection of a manufacturing process using low resin content prepreg material for fabrication of the remaining ACVF cover components. Process verification activity on the ribs is still in the tooling and development stage.

2.3.1 Cover Development

In the development of the low resin content material manufacturing process a series of flat panels were fabricated to evaluate bleeding arrangements. Evaluation of these panels indicated resin contents on the low side of the

TABLE 2-2. TEST PLAN

Layup	Static Tension (RTD)		Fatigue *	Σ
	Unnotched	Notched	Notched	
1	5	5	10	20
2	5	5	10	20
3	5	5	10	20
4	5	5	10	20
Σ	20	20	40	80

*Fatigue specimens will be preconditioned to 1% moisture content and tested to two lifetimes.

Layups: (1) $(\pm45/0/\mp45/\pm45)_s$ (3) $(\pm45/0_3)_s$

(2) $(\pm45/0/\mp45/\pm45/0)_s$ (4) $(0/+45/90/-45)_{2s}$

acceptable range of 26% - 32%. Therefore, additional panels were made to evaluate alternate bleeding systems. These panels included ply thickness variations and different bleeding arrangements as shown in Table 2-3. Based on the results shown in Table 2-3, the bleeding arrangement identified as number three was selected for further evaluation.

A series of hat stiffened panels were fabricated representing the root end section of the skin cover. Each panel contained three hat stiffeners 60 inches long. Panels were made both from standard (41%) and low (34%) resin content material. No prebleeding of any of these panels was performed prior to cure. The low resin content panels (identified as M25-3 and M25-5) used a stacking sequence similar to that illustrated in Figure 2-4.

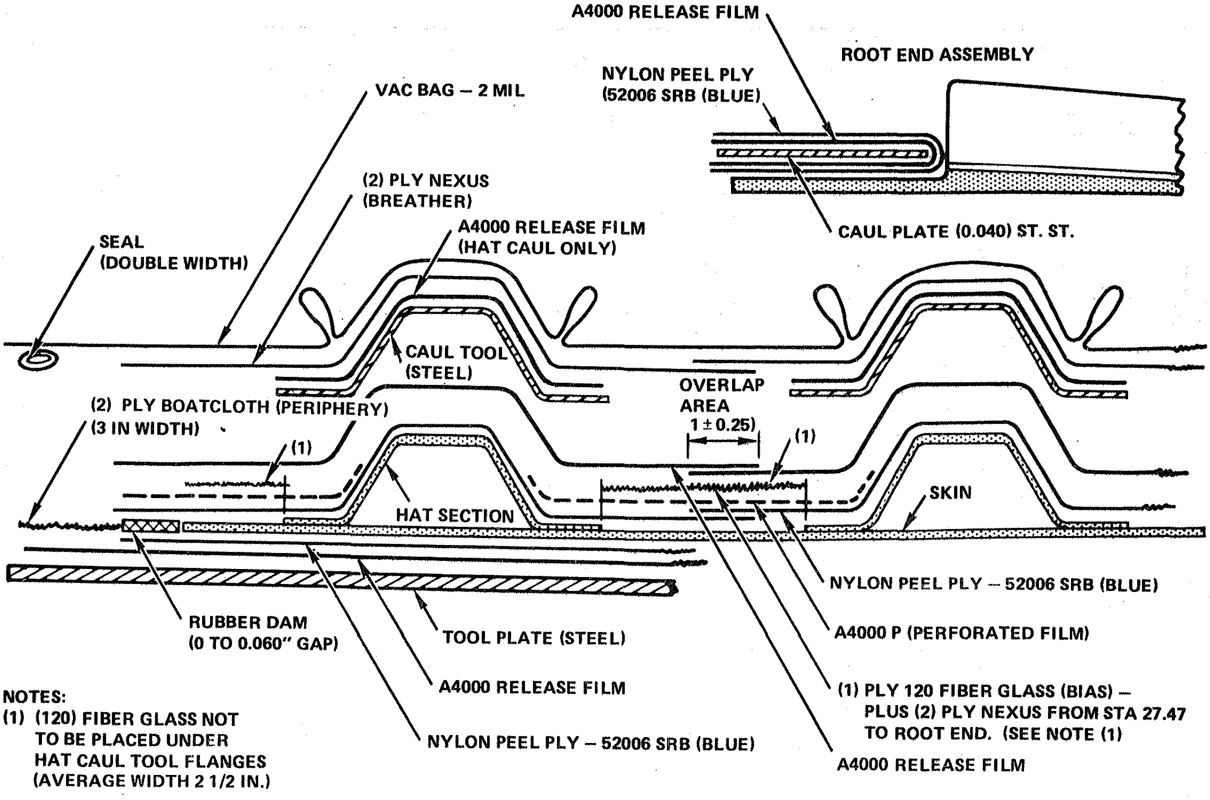
Two cure cycle alternates were carried into this final phase of cover development. These included the cure cycle developed for 41% resin content material without prebleed, and an alternate cure cycle which utilized less time at 260^{o}F and a faster heatup rate from 100^{o}F. These cure cycles are shown in Figure 2-5. The longer cure cycle was felt to offer improved volatile evacuation and assure temperature uniformity in the cover and supporting tooling.

A comparison of the mechanical properties of the three panels is shown in Table 2-4. All three panels produced results within acceptable limits, thus demonstrating that the properties required for ACVF covers could be satisfactorily achieved by use of either high or low resin content material. Fabrication of specimens for ancillary test was able to proceed on that basis with available (41%) resin content material.

Given the premise that no technical priority is applied between high or low resin prepreg material systems, a selection of the processing system for PRVT specimen fabrication and full-scale cover fabrication could be made based on producibility considerations. A comparison of the producibility aspects of cover fabrication was made and a recommendation to proceed using low resin prepreg material was adopted. A brief outline of the producibility factors considered and conclusions drawn is included in Table 2-5.

TABLE 2-3. RESIN CONTENT OF TRIAL PANELS LOW RESIN CONTENT PREPREG - ACVF

Panel No.	No. Plies	Bleeding Method No.	Bleeding Method Description		Resin Wt. %	Fiber Vol. %
			Bottom	Top		
1VL 1332 Flat	10	1	Laminate Nylon Peel Ply A4000 P3 Perf. Film Porous Armalon	Porous Armalon A4000 P3 Perf. Film Nylon Peel Ply Laminate	26.33	66.0
2VL 1332 Flat	10	2	Laminate Nylon Peel Ply Porous Armalon	Porous Armalon A4000 P3 Perf. Film Laminate	25.23	68.0
3VF 1332 Flat	16	1	Same as Described Above for Method 1		29.47	62.0
4VF 1332 Flat	16	3	Laminate Peel Ply A4000 P3 Perf. Film Porous Armalon	Porous Armalon A4000 P3 Perf. Film Laminate	30.92	61.0
1VL 1335 Tapered	(16)-(34)	1	Same as Described Above for Method 1		27.38	65.0
1VL 1337 Tapered	(16)-(34)	3	Same as Described Above for Method 3		30.4	62.0



NOTES:
(1) (120) FIBER GLASS NOT TO BE PLACED UNDER HAT CAUL TOOL FLANGES (AVERAGE WIDTH 2 1/2 IN.)

Figure 2-4. Bleeding System Assembly - Low Resin Content Prepreg

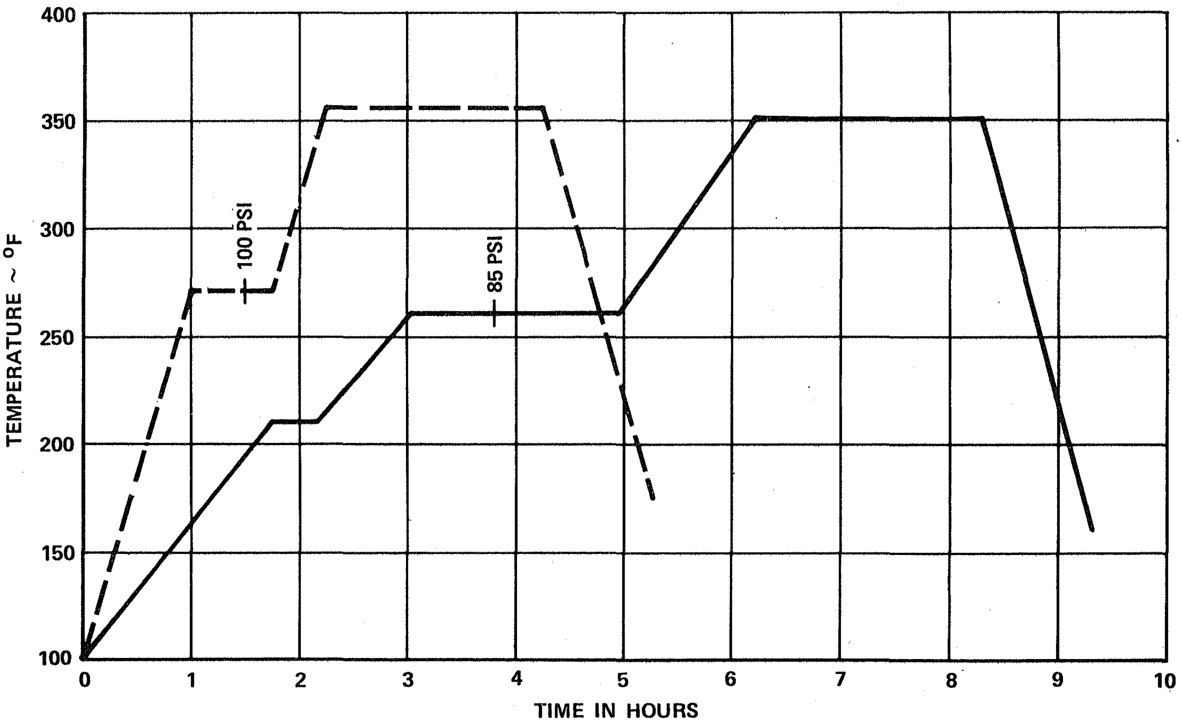


Figure 2-5. Cure Cycles

TABLE 2-4. PROPERTY AND PROCESS COMPARISON

Panel Number		M25-3	M-25-4	M-25-5
Material Process Description	Process Bulletin Requirement	High Resin No Bleed Long Cure Cycle	Low Resin Process Dev. Short Cure Cycle	Low Resin Long Cure Cycle
Resin Content %				
Leg	26-32%	29.1	30.3	28.6
Crown		26.9	28.3	27.6
Hat Pull Off Pounds		190	171	204
Flange				
SBS-RS				
RT. Dry	8	9.5	11.2	8.2
180°F Wet	7	8.9	10.3	8.2
Compression-KS1				
RT-Dry	69	91.8	88.6	96.5
180°F Wet	61	82.4	81.4	81.4
Crown				
SBS-RS1				
RT-Dry	9	9.7	13	9.4
180°F Wet	8	9.6	9.6	8.7
Compression-KS1				
RT-Dry	120	124.6	137.6	135.9
180°F Wet	100	117	112.9	111.7

2.3.2 Rib Development

Rib development during the past quarter has focused on three activities: 1) Methods of controlling the resin content in the cured part, 2) improvement and simplification of tooling, and 3) methods to improve the configuration and quality of the stiffening bead.

A series of seven truss ribs in the ancillary test configuration were molded during the reporting period. The first three of these were molded using the matched die concept and the remaining four were molded using an

TABLE 2-5. PRODUCIBILITY FACTORS - ALTERNATE NO BLEED FABRICATION SYSTEMS

Factor	Preference
1. Resin removal during cure	Low resin
2. Preparation of stack for cure	Low resin
3. Material handling, layup	No preference
4. Trim of uncured layup	Low resin
5. Control of resin flow	Low resin
6. Cure cycle	No preference
7. Tooling cleanup	Low resin
8. Trim cured laminate	Low resin
9. Part cleanup	Low resin
10. Tooling requirements	No preference
11. Repeatability confidence	Low resin

aluminum tool and a fiberglass caul plate. Also during this period a die was made to preform the 0-degree bead filler. The last four rib caps were fabricated with bead stiffeners made by this process. Resin content control was improved by varying the autoclave processing cycle, edge damming and using a peel ply to control resin flow.

Five actuator rib caps were also fabricated during this period. The first two of these were molded using a three-piece matched die concept. The second two were molded using a rubber block to define the bead side of the web with the aluminum portion of the tool to define the skin side contour and a flexible caul. The fifth actuator rib cap was molded using two pieces of the aluminum tool and a fiberglass caul plate on the side of the cap opposite the bead. A preformed bead was used in this cap.

The solid web rib ancillary tool, which is a two-piece matched die, was reworked to include flange joggles. A part was molded but lab tests revealed marginally low resin contents and physical properties.

The original tooling concept chosen for the truss and actuator rib caps and for the solid web rib was to use matched dies. All surfaces of the part would be configured by a machined aluminum tool. In theory, the sections could be brought together using either autoclave or press pressure resulting in parts with well controlled dimensions. During the Process Verification activities however, several deficiencies in this concept came to light. Most noticeably in the actuator rib cap, which is a three-piece tool, it was difficult to obtain uniform thicknesses in the flanges and webs, heat-up rates were low due to tool mass causing long autoclave cycles, assembly of the tool was awkward, and tooling costs would be high for a production program.

To alleviate some of these problems, investigation was begun into eliminating the top half of the die and substituting either a fiberglass caul or vacuum bag pressure only to replace the tool segment removed. Figure 2-6 illustrates the tooling setup for a truss rib cap using this concept. In another alternate simplify tooling, two actuator caps were molded using a rubber block to define the bead side of the web, the aluminum portion of the tool to define the skin side contour and a flexible caul. The rubber block and a part are shown in Figure 2-7.

Because of the large thermal coefficient of expansion of the rubber block, the problems associated with sizing the block, and heat-up control, this concept was not pursued any further.

An analysis of dimensions and physical properties of the rib specimens made with the 34% resin prepreg has shown that almost all have marginally low cured ply thicknesses. This has been corroborated by quality assurance laboratory tests which show low resin content and marginal physical properties. This indicates that the edge bleed which occurs during cure of the laminate is sufficient to reduce the resin content to an undesirable level. Two approaches were used to limit resin flow; the first was to use silicone rubber strips as a dam under the downward turning flange and the second to selectively use a peel ply to absorb and control the flow of resin. Also, modification of the cure cycle to a lower autoclave pressure and a shorter dwell period when the resin has low viscosity appears to be beneficial in retaining resin in the laminate.

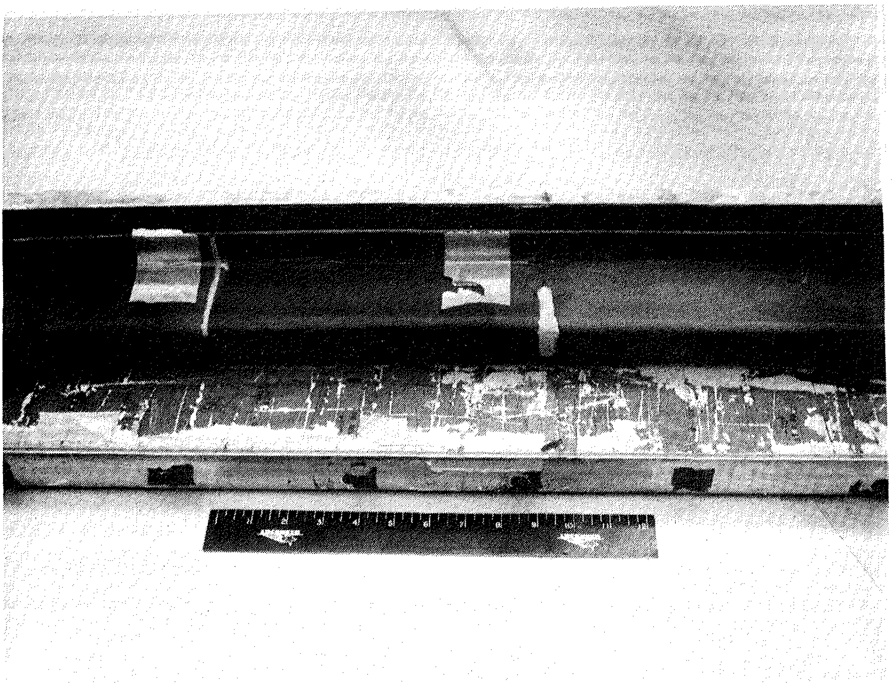


Figure 2-6. Fiberglass Caul Plate Used with Aluminum Tool to Mold Truss Rib Cap

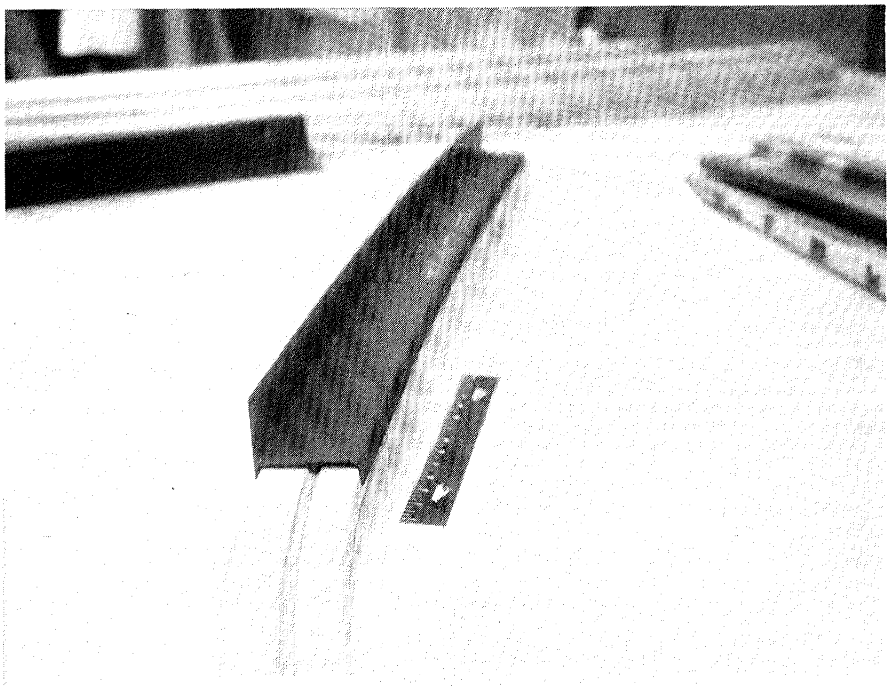


Figure 2-7. Rubber Block Tooling Concept

In summary of the process verification activity on rib caps, several conclusions may be drawn to form the basis for plans to complete rib development activity:

- As evidenced by results of truss rib caps made by replacing one portion of the aluminum tool with a carefully made fiberglass caul plate, such caul plates would be satisfactory for the full-size components. Tool economies would be realized by eliminating the machining of that portion of the tool. Further, tool mass is reduced, allowing greater flexibility in selecting heat-up rates.
- A recent specimen was molded which made use of a silicone rubber dam under the downturning flange to prevent runout of the resin during that period of the cure cycle in which the resin has a low viscosity. This plus reduction of autoclave pressure and a shortened pregellation dwell, also appears to reduce resin runout which in the past has caused low resin content and degraded physical properties.
- It is uncertain whether satisfactory stiffening beads can be made to the existing design. As a result, investigation will be conducted into alternate approaches to stiffening the rib caps.

2.4 CONCEPT VERIFICATION

The concept verification tests are designed to verify the structural integrity of the most critical areas of the fin box structure. Tests include static and spectrum fatigue under various temperature and moisture conditions. The status and results of these tests are summarized in Table 2-6.

The structural integrity of the cover design was verified with the successful test of a test specimen which was representative of the cover-to-fuselage joint.

2.4.1 Surface Attachment to Fuselage - Test Item H25

The objectives of this test were:

1. To verify the static-compression strength of the cover assembly.
2. To substantiate predicted analytical buckling results with a strain gage and moire' shadow analysis.

The excellent agreement between the predicted and the actual test results is illustrated in Table 2-7.

TABLE 2-6. STATUS OF CONCEPT VERIFICATION TESTS

Test No.	Test Description	Type Test/ Condition	Test Status/Results
Covers			
H-25	Surface Attach to Fuselage	Static - Compression RT Dry	Test Complete - Failed at 141% of Design Ultimate Load
H-26A	Stiffener Runout	Static - Tension Wet - Temp Cycled	Test Specimen Fabrication in Progress
H-26B	Stiffener Runout	Fatigue - 2 Lifetimes RT Dry	Test Specimen Fabrication in Progress
H-27	Surface Panel Stability	Static - Compression - Elevated Temp. Wet	Test Setup and Specimen Complete
H-28	Surface Panel Fail Safety	Fatigue for 1/2 Lifetimes RT Dry	Test Setup and Specimen Nearing Completion
H-29	Lightning Strike	RT Dry	Test Specimen Fabrication in Progress
Ribs			
H-24AT	Rudder Hinge Ftg. - Truss Rib	Static - RT Dry	Test Specimens in Process Development Stage
H-24AS	Rudder Hinge Ftg. - Solid Web	Static - RT Dry	Test Specimens in Process Development Stage
H-24B1	Actuator Ftg. to Web Attachment	Static - RT Dry	Test Specimens in Process Development Stage
H-24B2	Actuator Ftg. Web Attachment	Fatigue - RT Dry	Test Specimens in Process Development Stage
H-24C	VSS 97.19 Rib	Static - RT Dry	Test Specimens in Process Development Stage
H-20A	Rib Beam Cap	Static - RT Dry	Test Specimens in Process Development Stage
Spars			
H-20B	Spar Beam	Static - RT Dry	Test Complete - Failed at 163% of Design Ultimate Load
H-20B	Spar Beam	Static - Elevated Temp. Wet	Test Complete - Failed at 181% of Design Ultimate Load
H-21A1	Spar Web	Static - RT Dry	Test Complete - Failed at 113% of Design Ultimate Load
H-21A2	Spar Web	Static - Thermo Cycled	Test Complete - Failed at 123% of Design Ultimate Load
H-23	Spar Joint	Static - RT Dry	Test Complete - Failed at 129% of Design Ultimate Load

TABLE 2-7. H-25 PREDICTED VERSUS ACTUAL TEST RESULTS

Event	Predicted	Actual
Onset of Panel Buckling	No prediction	Between 30,000 and 40,000 lb
Panel Buckling (Fully developed buckling mode shape)	48,000 lb - Based on NASA's VIPASA buckling program	Moiré fringe pattern analysis is presently being interpreted - Approximately 47,000 to 48,000 lbs based on strain gauge data
Panel Failure	78,000 lb	81,900 lb (141% Design Ultimate Load)
Failure Location	16-ply laminate between VSS 97.19 and VSS 121.45	16-ply laminate adjacent to VSS 121.45 Rib Station

2.4.1.1 Test Setup and Specimen Preparation

The panel test setup is shown in Figures 2-8 and 2-9. As shown in Figure 2-8, a rigid steel reaction frame was used in order to help stabilize the test panel during compression loading. The panel was prevented from buckling at the two rib supports and at the root end tee through the use of three 27-inch long aluminum alloy flexure plates. These plates were designed to provide an end restraint coefficient of approximately 1.0 at the test panel. Kick loads were reacted through four solid steel links attached to the ends of the panel assembly (at the centroids) and to the reaction frame.

Prior to installing the test panel in the 400-kip static test machine, four axial strain gages and two T-gages were applied directly to the composite material. The gages were located back-to-back and were connected to read as eight separate gages, six axial and two transverse.

In addition to the above strain gages, four linear variable differential transducer (LVDT) deflection indicators were positioned between the upper compression head and the movable base. These deflection indicators, which

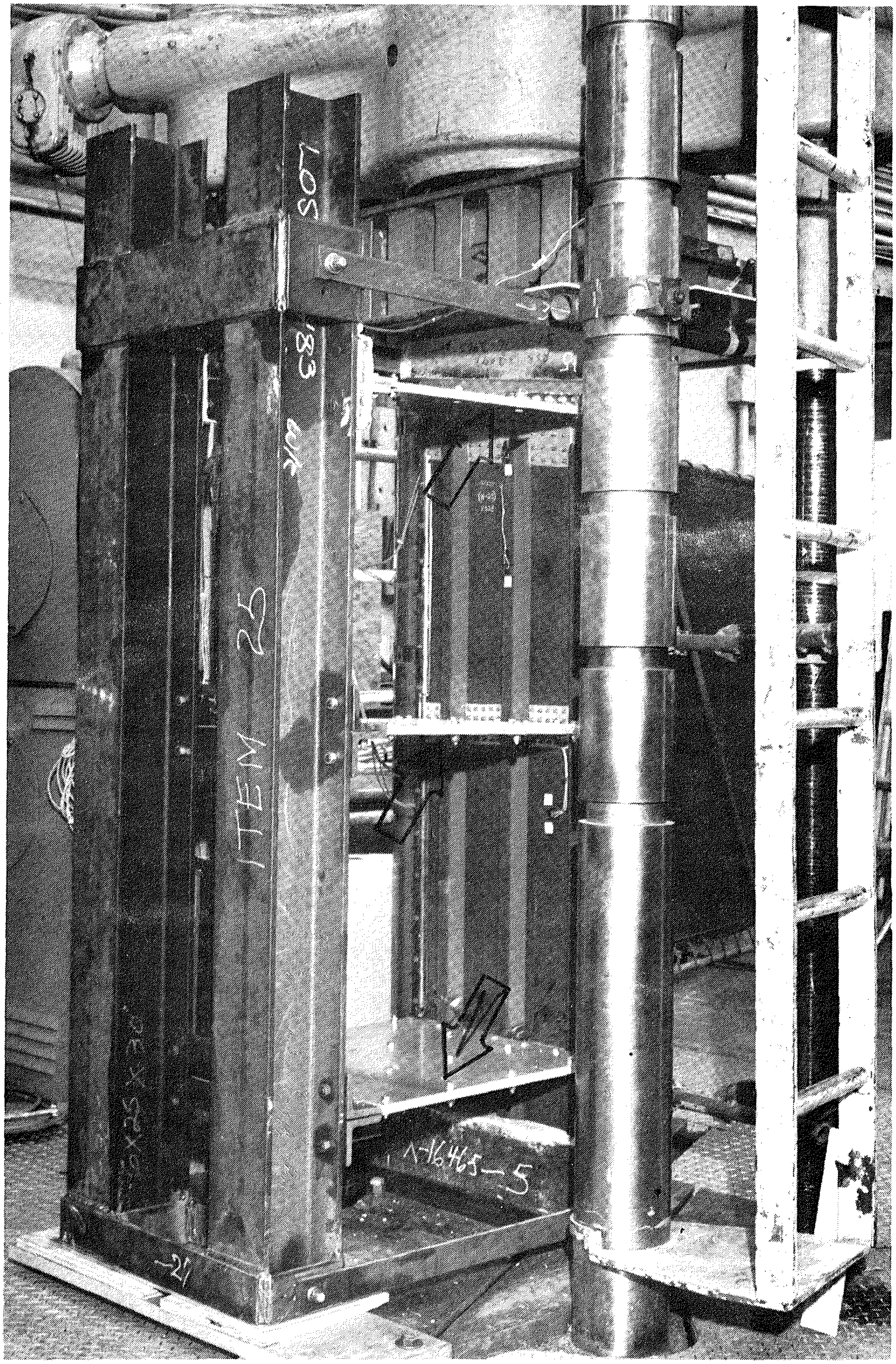


Figure 2-8. Panel in Fixture Showing the Three Flexures in Place

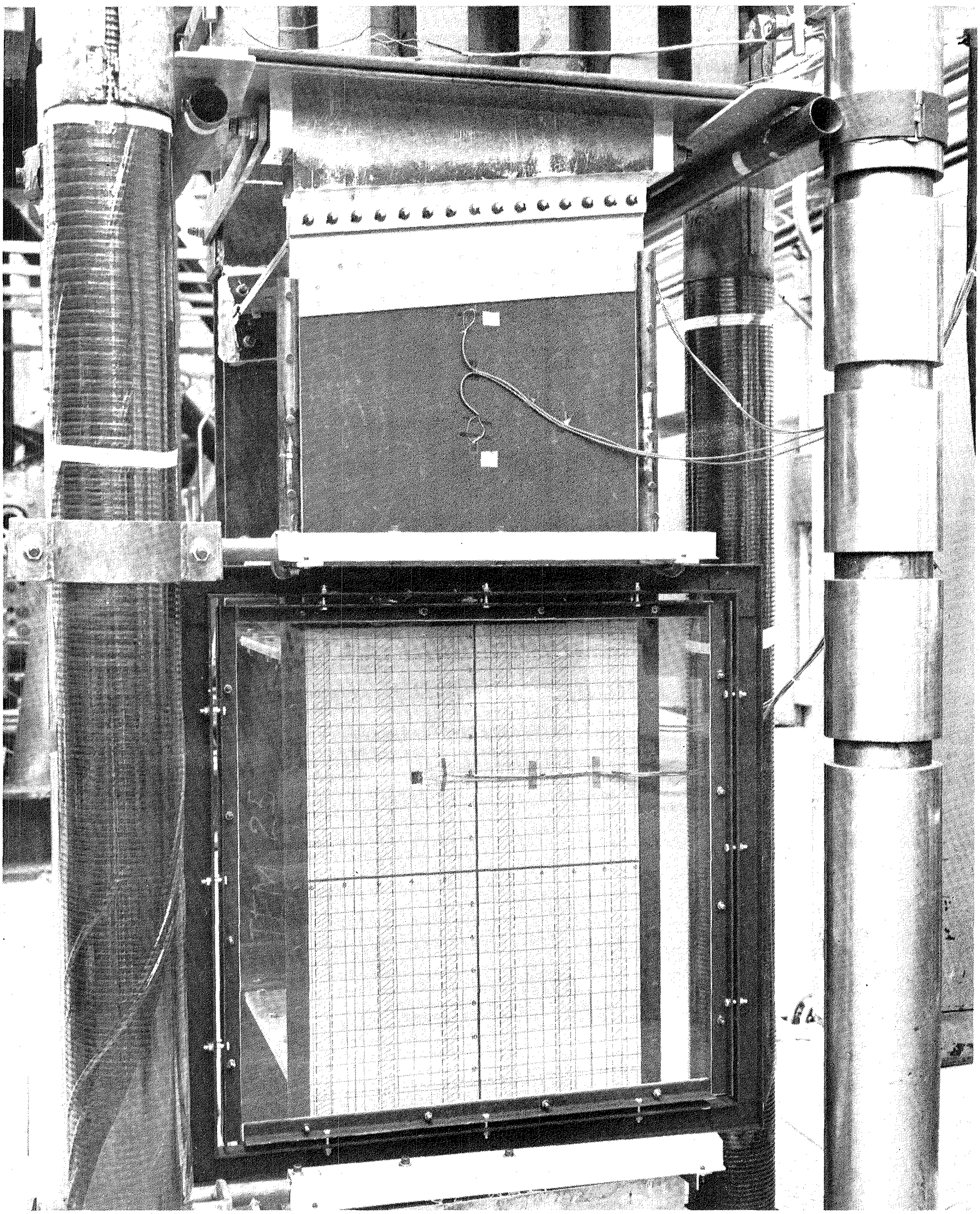


Figure 2-9. H25 Panel in Test Setup Showing the Moiré Grid in Place

were arranged symmetrically about the upper compression head, were used as an aid in the initial alignment of the upper compression head with the lower compression plate. This provides a continuous record of the incremental changes in panel length, and also provides an indication of any load redistribution which might occur prior to failure.

The outside surface of the panel opposite the center bay (i.e., the area between the two rib supports) was prepared for shadow moiré. This was done first by spraying the panel with a flat white enamel in order to provide the optimum matte surface for fringe enhancement. A one-inch coordinate grid system was then ruled over the painted surface. In addition to these grid lines, the center lines of the three stringers, the two rib support stations, and the contact surfaces between the stringer flanges and the skin were identified. After the panel was located in the test machine, a moire' grid (containing 50 vertical lines/inch at 40% density) was positioned 1.00-inch away from the painted surface as shown in Figure 2-9. A 1200-watt carrousel projector, with a lens cap containing a 1/16-inch vertical slot, was used as the light source.

2.4.1.2 Panel Test

The panel was tested dry at room temperature and was loaded three times prior to the final test. Below is a summary of these loadings:

(1)	0-12 kips-0	to check gage polarity and initial slope of LVDT-generated deflection curves.
(2)	0-30 kips-0	to check out shadow moiré setup and to determine secondary slope of deflection curves.
(3)	0-57.5 kips-10	load increased to design ultimate, in 10-kip increments, then reduced to 10 kips (data, movies, and shadow moire' photos taken).
(4)	10-81.91 kips	panel loaded to failure at approximately 3 kips/second (data, movies, and shadow moire' photos taken).

The panel failed in compression at 81.91 kips and is shown in Figures 2-10 and 2-11. A high speed motion picture indicated that the failure began when hat stiffener No. 3 pulled loose from the skin.

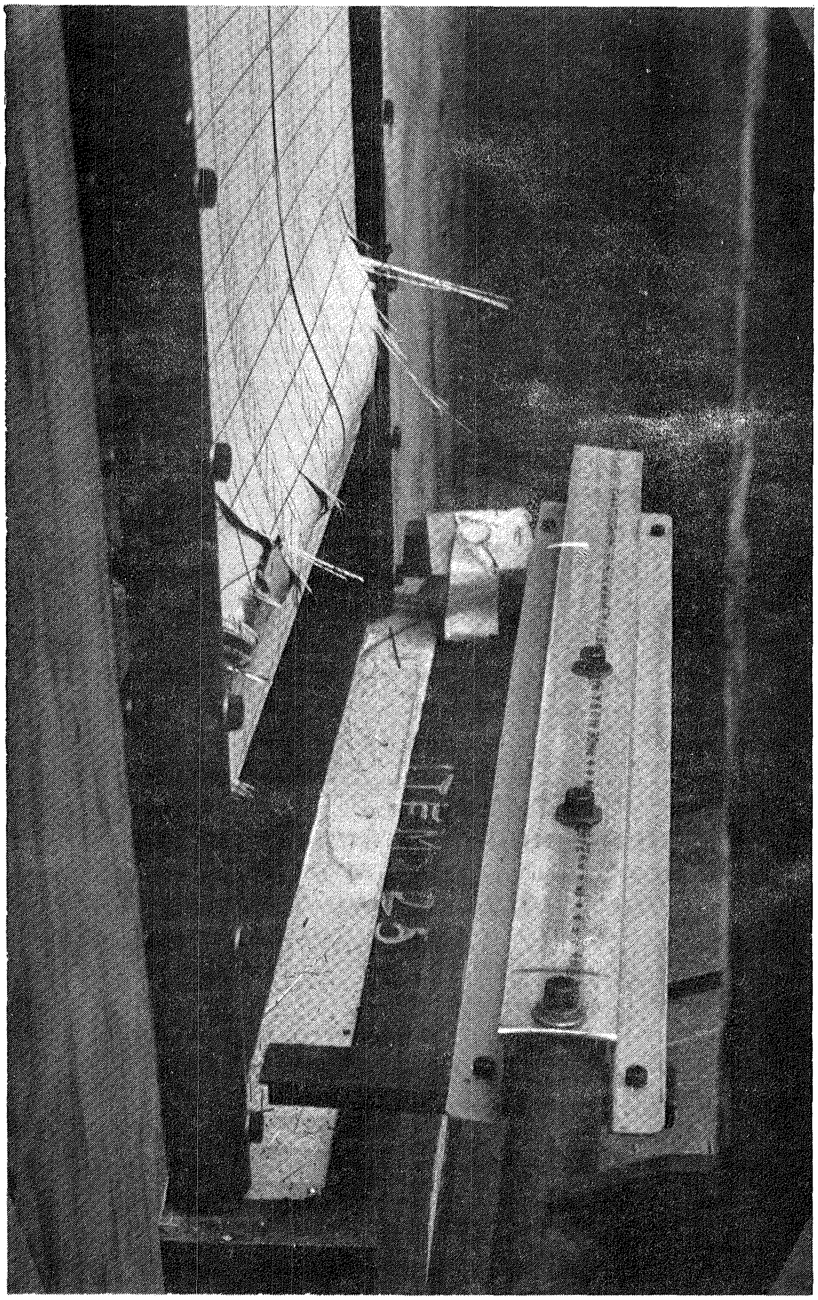


Figure 2-10. Nature of the Failed Panel on the Moiré Grid (Flat) Side

Figure 2-11. Nature of Failure Throughout the Hat Stiffeners

Another important objective that was satisfied by the H-25 test was the verification of the static strength of the root end joint. Even though the panel test was not designed to interrogate the full-strength of the root end, the joint and adjacent laminate were subjected to 106% of its design ultimate load without evidence of incipient failure.

In addition to the VIPASA buckling analysis program used to study critical instability behavior of individual panel elements, it was also necessary to analyze the test panel as a beam-column in order to properly understand the load/strain data.

Figure 2-12 compares the measured H-25 test strains versus the predicted strains derived from the beam-column analysis. An axial force of 40,000 pounds was selected for the analysis because it was believed to represent the load just prior to the onset of buckling, yet sufficiently high enough to produce measurable beam-column behavior.

The measured strains (depicted by circles) generally agreed with beam-column results except at the following locations:

- Stringer Runout (Gage No. 1) - In the analysis, the hat crown area adjacent to gage No. 1 was conservatively assumed to remain 50% effective; however, actual strain measurements verify that only 2 - 3% of the peak crown load remains in the hat stiffener at gage location No. 1. Strain measurements from gage No. 2 located on the skin showed excellent agreement with predicted strains. The load-strain behavior of gages No. 1 and No. 2 are shown in Figure 2-13.
- Station 34.92 (Gage locations No. 5 and No. 6) - Transverse as well as axial gages at this location showed that at 40,000 pounds some local skin buckling was developing between the hats producing both higher (no. 6) and lower (No. 5) measured strains than predicted in the beam-column study.

Figure 2-14 shows the measured strains for axial gages No. 5 and No. 6. Although strain reversal is displayed between these two back-to-back gages beginning at 30 to 40 Kips, the inflection points occurring in both traces at a load of 47 to 48 Kips are undoubtedly a more realistic indication of the point of fully matured buckling. Also shown for reference in Figure 2-14 is a point representing the NASTRAN 3-D model strain for the 16-ply skin laminate

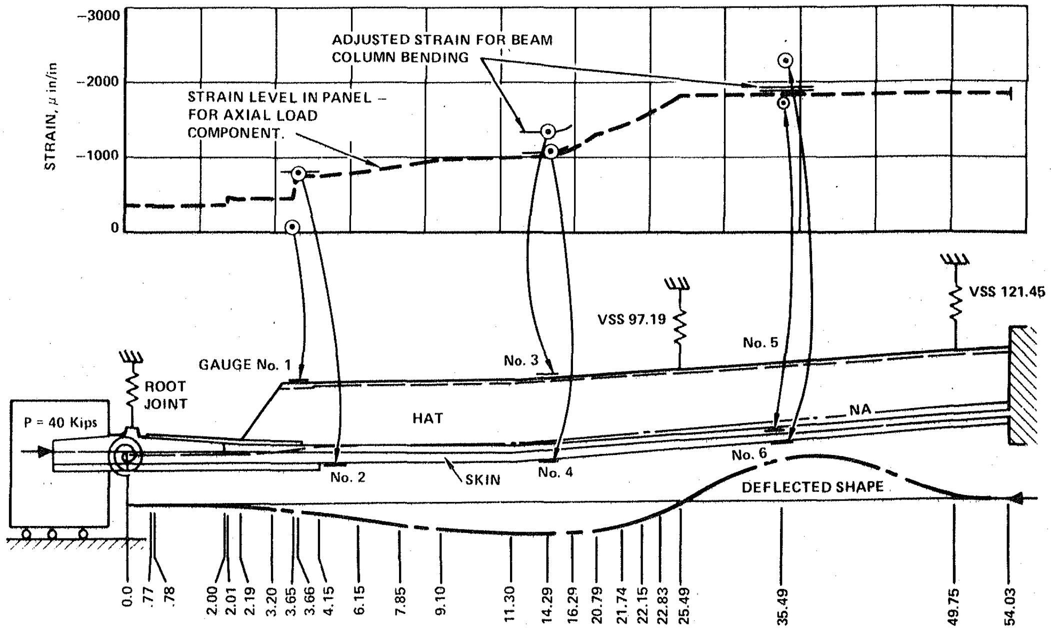


Figure 2-12. H-25 Predicted Beam-Column Strains vs Measured Strains

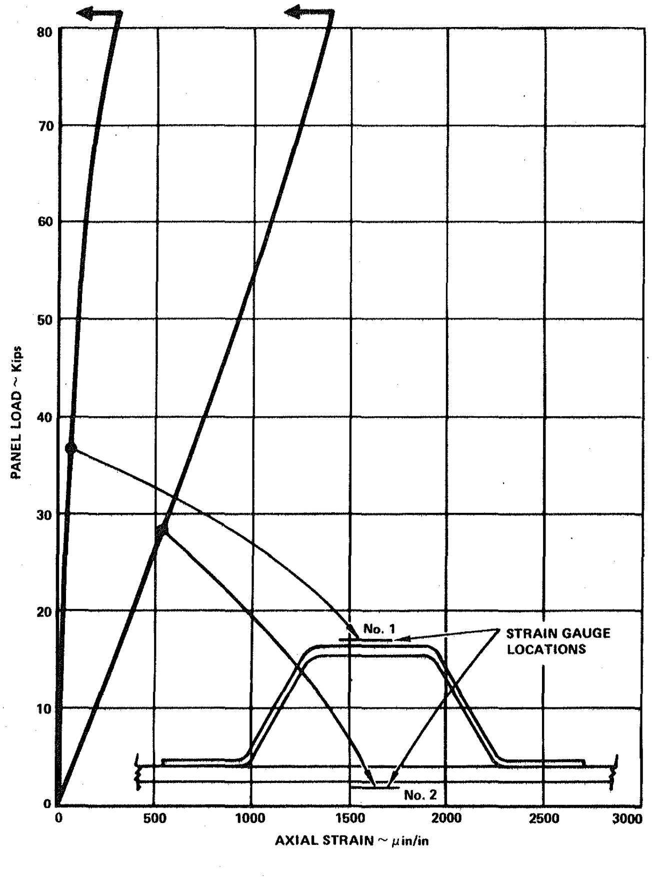


Figure 2-13. H-25 Measured Load-Strain Behavior at Stringer Runout

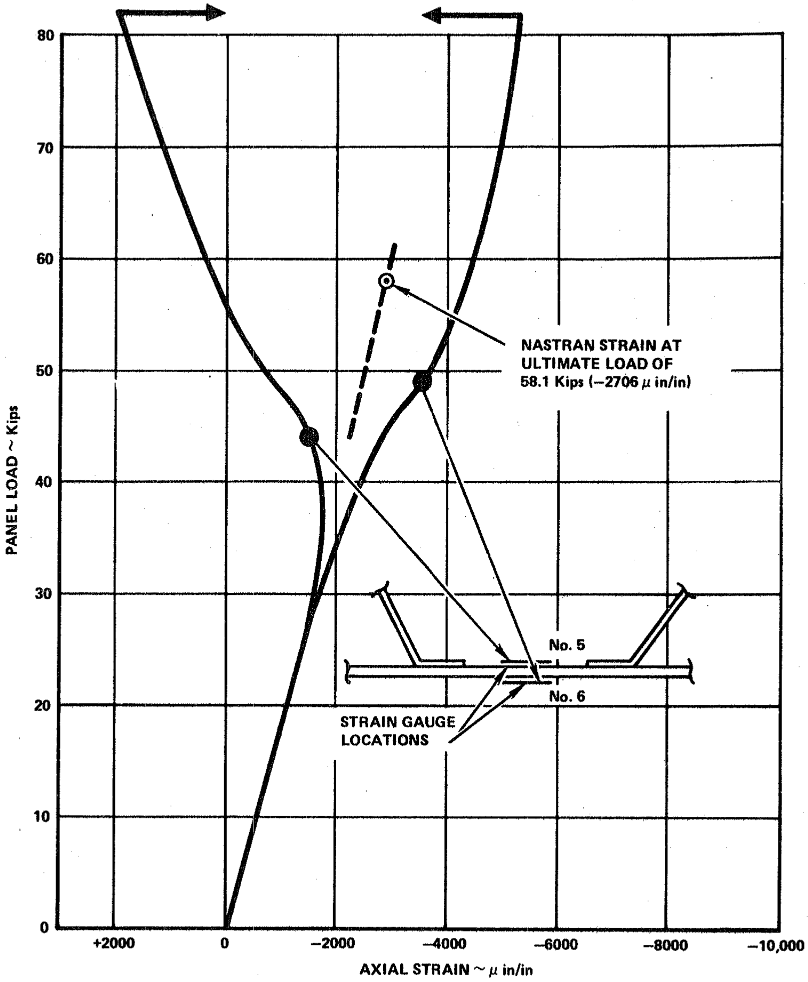


Figure 2-14. H-25 Measured Load-Strain Behavior at Sta. 34.92 in.

corresponding to this station. Note that if bending and local buckling were not influencing the shape of these traces, the initial linear elastic slope would appear to intercept the NASTRAN predicted strain at a design ultimate load of 58.1 Kips. The overall configuration and corresponding NASTRAN ultimate loads for the panel are shown in Figure 2-15.

Figure 2-16 shows the measured load-strain behavior in the skin and hat crown at a distance of 13.5 inches from the root joint. At this station there is excellent agreement between the measured and predicted beam-column strains.

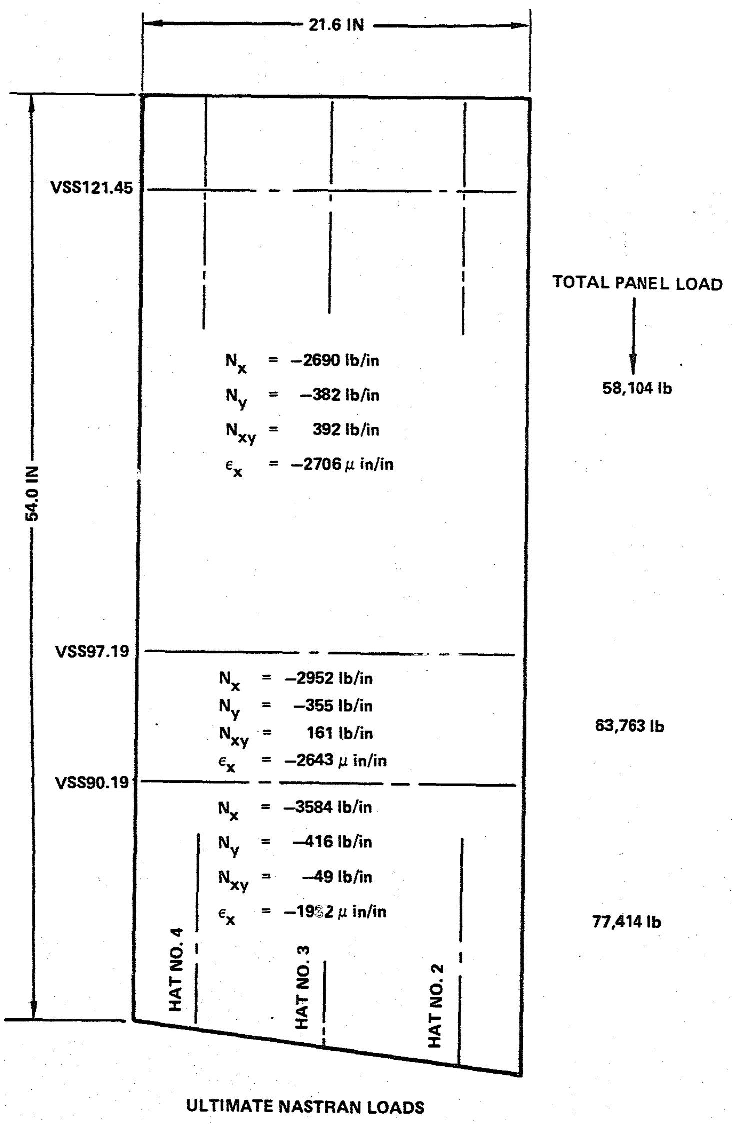


Figure 2-15. H-25 Test Specimen Configuration and Associated Design Loads

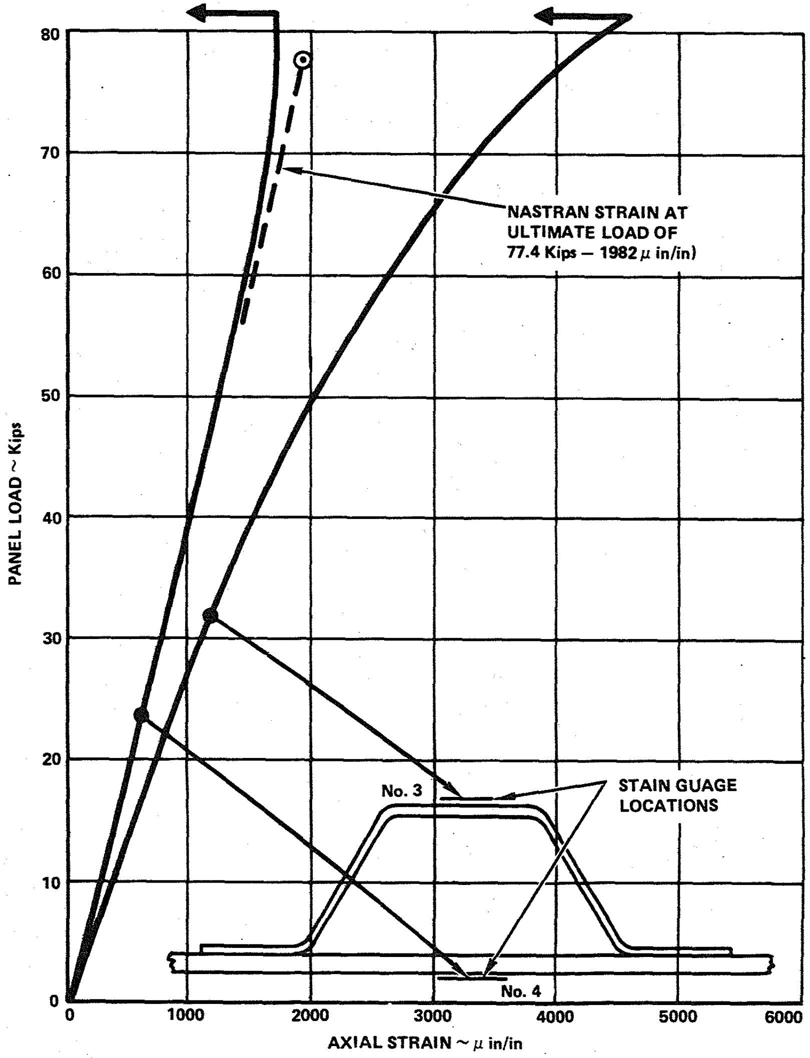


Figure 2-16. H-25 Measured Load-Strain Behavior at Sta. 13.5 inches

Structural analysis based upon these strain measurements suggests that at this station the hat crown is carrying its predicted share of the total panel load (12.5%).

It should be noted that at this station in the test panel there is no visual indication of local buckling occurring even at the failure load of 81,900 pounds. The curvature in these traces is created solely by the column bending. Reflected in Figure 2-16 is the NASTRAN strain at ultimate load for this location and again, as in Figure 2-14, good correlation exists with the measured strain data from the H-25 test.

2.4.2 H-27 Surface Panel Stability

The objective of this test is to verify the stability of the cover structure at elevated temperature and with 1% moisture absorption. The specimen (setup) has been completed and is shown in Figure 2-17. The test will be accomplished immediately after the first of the year.

2.4.3 H-28 Surface Panel Fail Safety

The objective of this test is to establish the crack propagation rates and fail-safe features of the hat-stiffener cover design. Figure 2-18 shows the panel in the composites lab having the fiberglass end reinforcement laid up. Testing is scheduled to commence in February 1979.

2.5 QUALITY ASSURANCE

2.5.1 Laboratory Activities

The Quality Assurance Laboratory continued to perform two basic functions during the reporting period: (1) Batch testing of T300/5208 to ensure that the graphite/epoxy material is acceptable prior to its use, and (2) testing of parts fabricated for either the process development studies or the Engineering Ancillary Test Program.

2.5.1.1 Acceptance Tests

Three high resin content and three low resin content material batches were tested for uniformity during this reporting period. All material batches

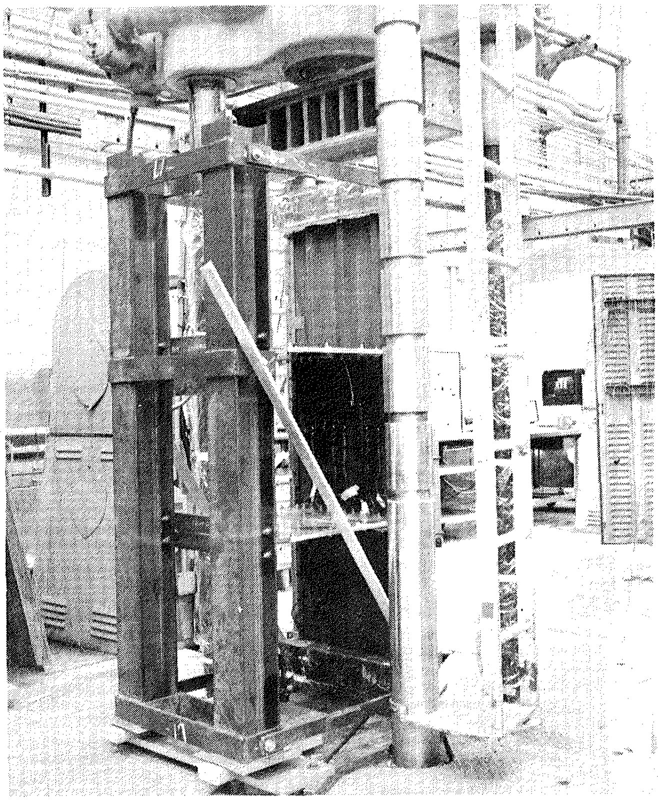


Figure 2-17. Stability Panel Test Setup

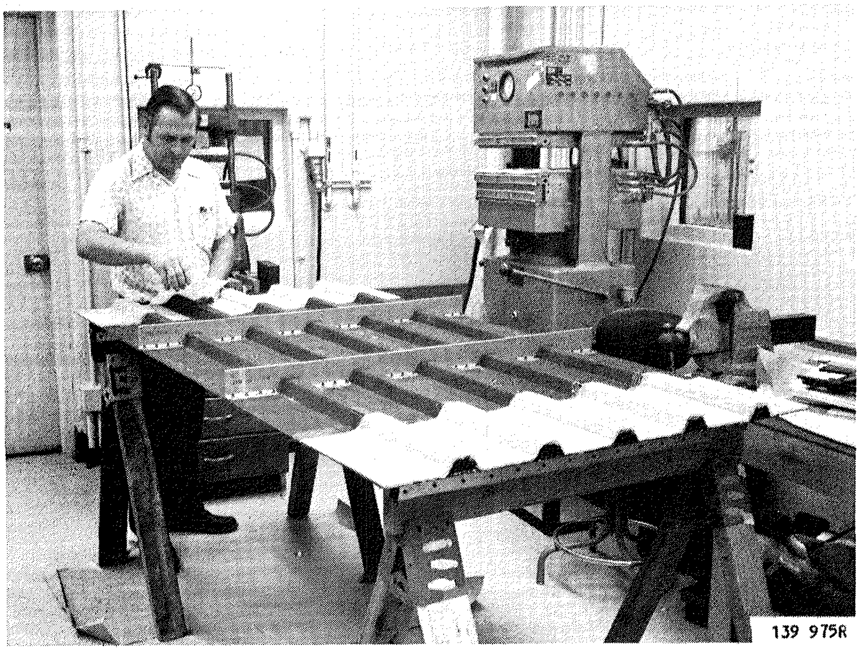


Figure 2-18. Surface Panel Fail Safety

were accepted for use. The test results for a low resin material batch are shown in Table 2-8.

2.5.1.2 Process Development Tests

Quantities of hat/skin assemblies, flat panels, and rib caps were tested in support of the single-stage cover process development and rib process development programs. The results of these tests are given in Section 2.3. Specific tests include compressive strength, short beam shear, hardness, resin content and specific gravity. In addition, numerous photomicrographs were taken for correlation with laboratory test data and NDI.

2.5.2 Inspection Activities

In-process inspection is provided on all ancillary test specimens, on PRVT components, and on all full-scale vertical fin components. Laminates are inspected during layup for proper positioning and fit in the tool, correct number of plies, ply orientation, gaps between segments of prepreg tape, absence of contaminants, and other visual anomalies such as wrinkles and improper fit-up.

The various processing operations during fabrication are under a continuing quality surveillance for compliance to the approved sequence and methods required to provide components that meet engineering design parameters. Inspection ensures that perishable materials are properly identified and used according to specification requirements within the allowable shelf life.

Cured assemblies are inspected visually for surface defects, blisters, excess resin deposits, resin-starved areas, pits, cracks, voids, and other surface discontinuities. Thickness measurements are made at selected points for correlation with laboratory tests.

The FAA Designated Manufacturing Inspection Representative (DMIR) and NAVPRO inspectors participate in the inspection of all ancillary test specimens and PRVT components. The following Conformity Inspections were completed during the current reporting period:

TABLE 2-8. T300/5208 BATCH ACCEPTANCE TEST RESULTS

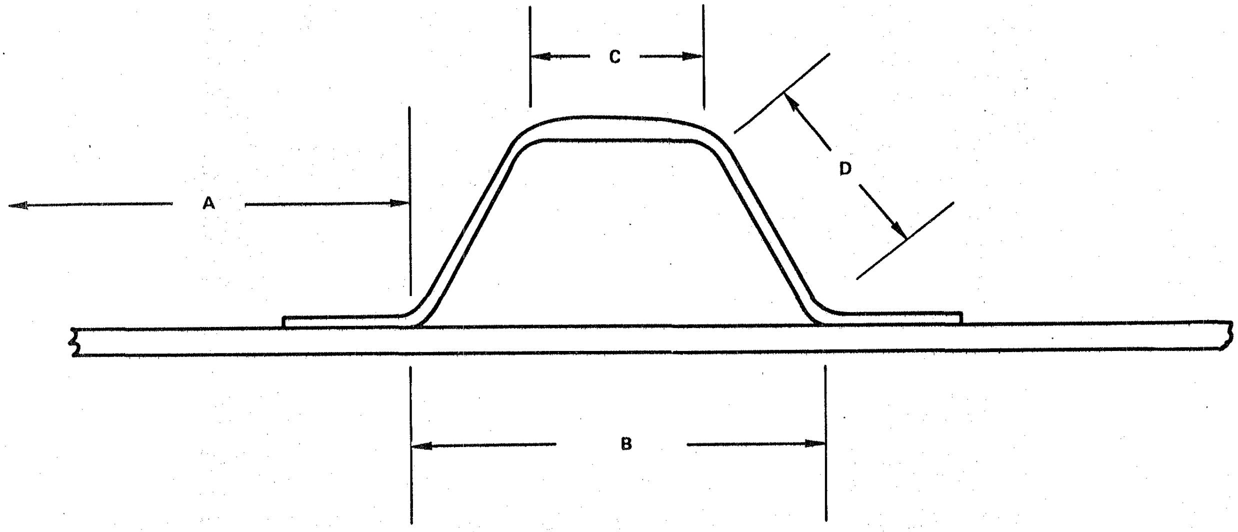
Preliminary Draft #2 (10-23-78) C-22-1379A/114 Specification Requirements		Results of Test					
		1	2	3	4	5	Avg
Areal wt (4) (3" x 3")	139-149 gms/meter^2	146	148	147	147		147
Liquid chromatography (1)							
Volatiles (2) (60 ±5 minutes at 350°F)	3.0 max edge center	.30 .27					
Dry resin content (4) (3 X 3")	31-37%	36.3	35.6	36.3	36.4		36.2
Flow (2) at 350°F at 85 psi	9-18%	15.3	15.0				
Gel time (2) at 350°F	Info only, minutes	19.3	19.2				
Cured fiber volume (3) 0.080 in. panel	60-68%	61.6	61.6	62.0			61.7
Cured fiber volume (3) 0.040 in. panel	60-68%	62.4	62.5	62.0			62.3
Specific gravity (3) 0.080 in. panel	1.54-1.60	1.561	1.563	1.562			1.562
Specific gravity (3) 0.040 in. panel	1.54-1.60	1.563	1.562	1.561			1.562
Tensile strength, longitudinal (3) at 75°F	190 ksi, min, ind	241	237	227			235
Tensile mod., longit. (3) at 75°F (per Fig. 1)	18.5 X 10^6 psi, min, ind	21.2	21.8	20.5			21.2
Flexural strength (3) at 75°F	210 ksi, min, ind	260	271	280			270
Flexural modulus (3) at 75°F	18 X 10^6 psi, min, ind	18.3	18.4	19.4			18.7
Flexural strength (3) at 180°F	200 ksi, min, ind	211	216	243			224
Flexural modulus (3) at 180°F	16 X 10^6 psi, min, ind	17.7	18.3	17.6			17.9
Short beam shear (3) at 75°F	13 ksi, min, ind	18.8	19.6	19.2			19.2
Short beam shear (3) at 180°F	12 ksi, min, ind	15.3	14.7	15.5			15.2
Thickness per ply (5) 0.080 in. panel	.0046-.0056 in.	.0051	.0051	.0050	.0050	.0051	.0051
Thickness per ply (5) 0.040 in. panel	.0046-.0056 in.	.0054	.0054	.0054	.0053	.0053	.0054

Notes: Batch 1237
Date 11-14-78
Lab Report 351735

H12A, 12A-1, 12A-5 & 12A-7	Impact Test Panels
H25	Surface-to-Fuselage Joint Specimen
H27	Surface Panel - Stability
H28	Surface Panel - Fail Safe

2.5.2.1 Nondestructive Inspection (NDI)

During this reporting period, activity has centered on support of manufacturing activities. All hat-stiffened panels and ribs are being ultrasonically inspected one hundred percent. The ultrasonic inspection techniques used on the hat stiffened panels are shown in Figure 2-19. This process becomes somewhat involved when dealing with H25 hat stiffened specimens due to the large number of thickness changes. A detailed ultrasonic inspection procedure was written to aid in the inspection of these complex assemblies. Figures 2-20 through 2-21 provide an example of the ultrasonic C-scan results obtained on an H12A (Damage Tolerance) ancillary test panel.



A = REFLECTED THRU TRANSMISSION: MULTIPLE GAIN SETTINGS FOR THE VARIOUS THICKNESSES INVOLVED 5 MHz TRANSDUCER

B = PULSE-ECHO: GATED ON BACKSURFACE 10 MHz FOCUSED TRANSDUCER

C = PULSE-ECHO: SAME AS B

D = PULSE-ECHO: HANDSCAN CONTACT 10 MHz TRANSDUCER

Figure 2-19. Inspection Techniques for Cover Panels

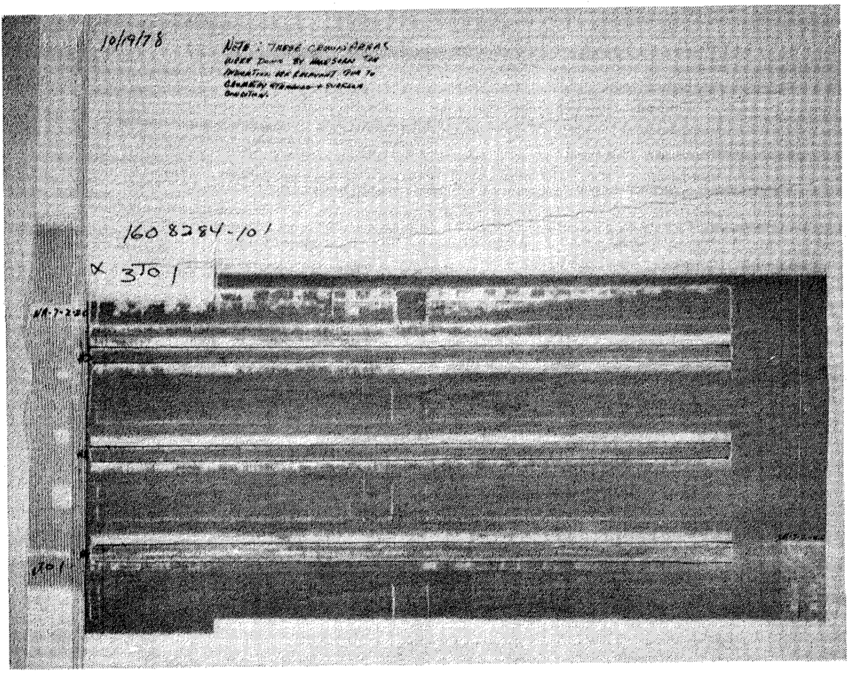


Figure 2-20. Ultrasonic C-Scan H-12A-1 Crown Areas

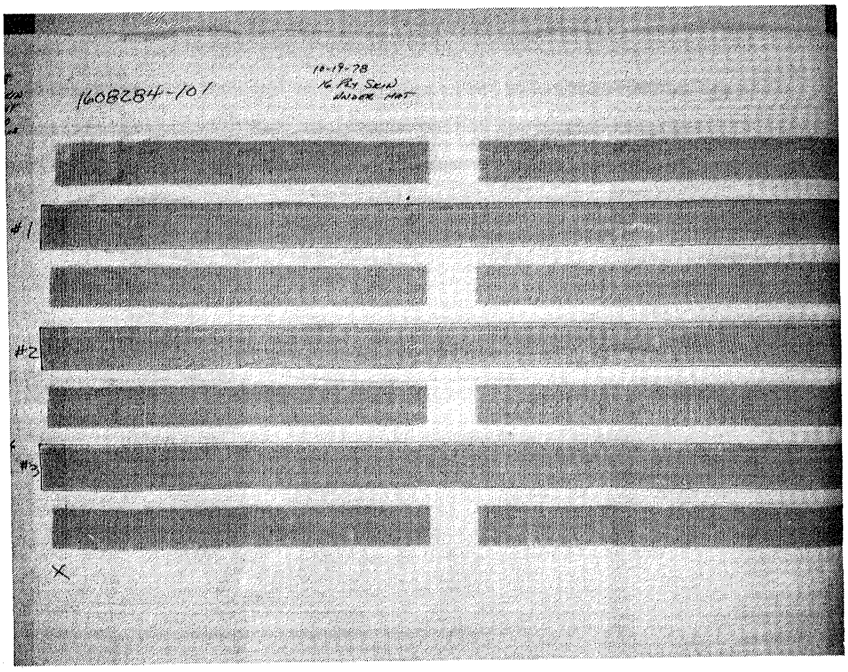


Figure 2-21. Ultrasonic C-Scan H-12A-1 16-Ply Area Under Hat

SECTION 3

PHASE III - PRODUCTION READINESS VERIFICATION TESTS

The ACVF program does not include flight service evaluation but alternately provides for multiple large-scale subcomponents of the structure for evaluation of variability in static strength and for assessment of durability under extended-time laboratory tests involving both load and environment simulation. The production readiness verification program (PRVT) is supplemental to the ancillary test program. These tests are designed to provide information to answer the following questions:

- What is the range of production qualities that can be expected for components manufactured under conditions similar to those expected in production, and how realistic and effective are proposed quality levels and quality control procedures?
- What variability in static strength can be expected for production quality components, and are the margins sufficient to account for this variability?
- Will production quality components survive extended time laboratory fatigue tests involving both load and environment simulation of sufficient duration and severity to provide confidence of in-service durability?

The questions are not primarily directed towards basic material properties. It is believed that the combination of service experience on secondary structures and coupon tests in the ancillary test program provide confidence in durability of the basic material. The questions are directed instead to the realities of production quality as influenced by cost objectives and by scale-up and complexity effects which will cause structural quality to differ from that represented by idealized small coupons.

On each of two key structural elements of the ACVF, ten static-strength tests and ten durability tests will be conducted. One element will represent the front spar/fuselage attachment area, and the other element will represent the cover/fuselage joint area.

3.1 FACILITY PREPARATION

During the last quarter the spar and cover environmental chambers have been delivered and installed in the load reaction frames. Figures 3-1 and 3-2 show one of the two spar chambers mounted on the load reaction frame. The installation of the spars in the environmental chamber is shown in Figure 3-3.

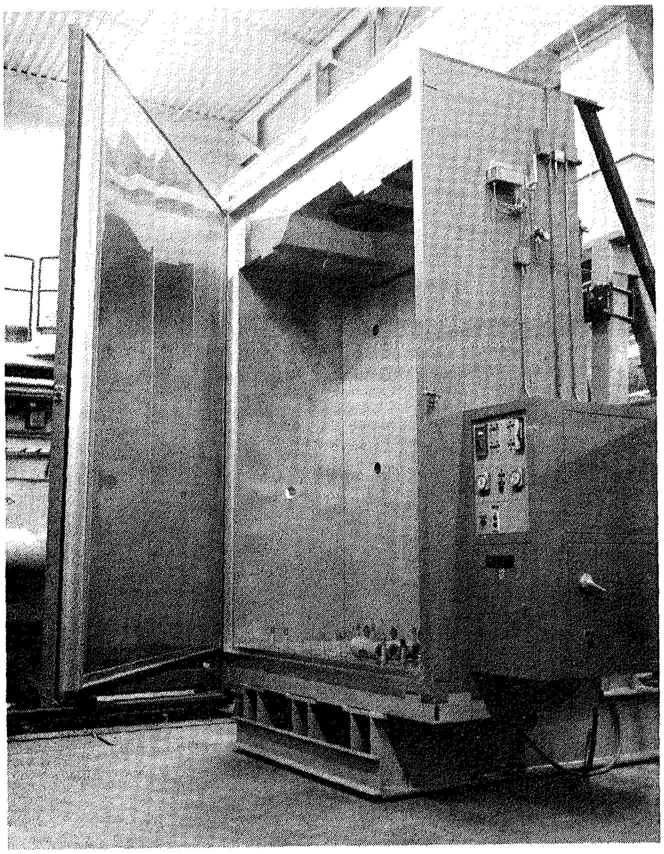


Figure 3-1. Spar Test Chamber Mounted on Load Reaction Frame

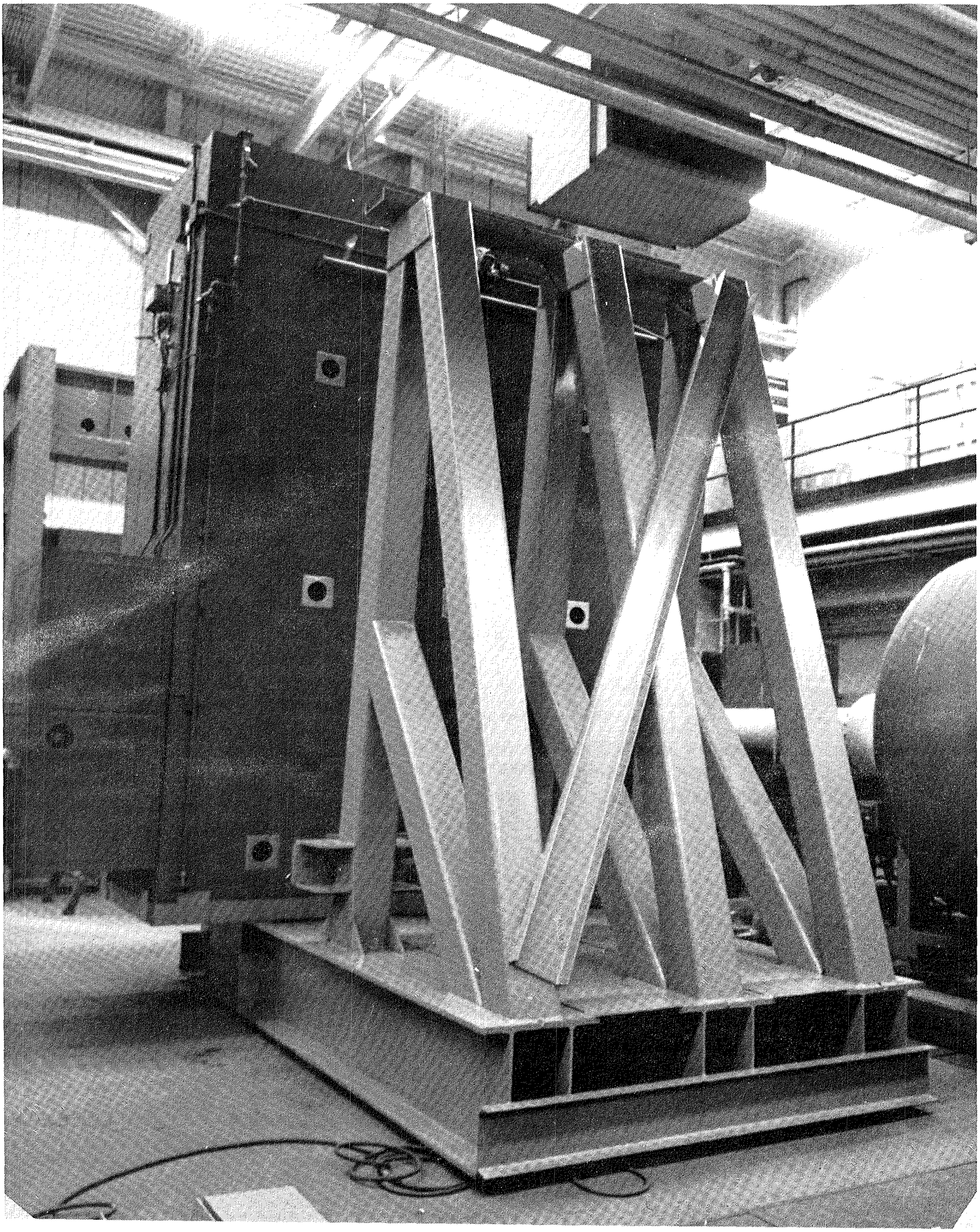


Figure 3-2. Spar Load Reaction Frame

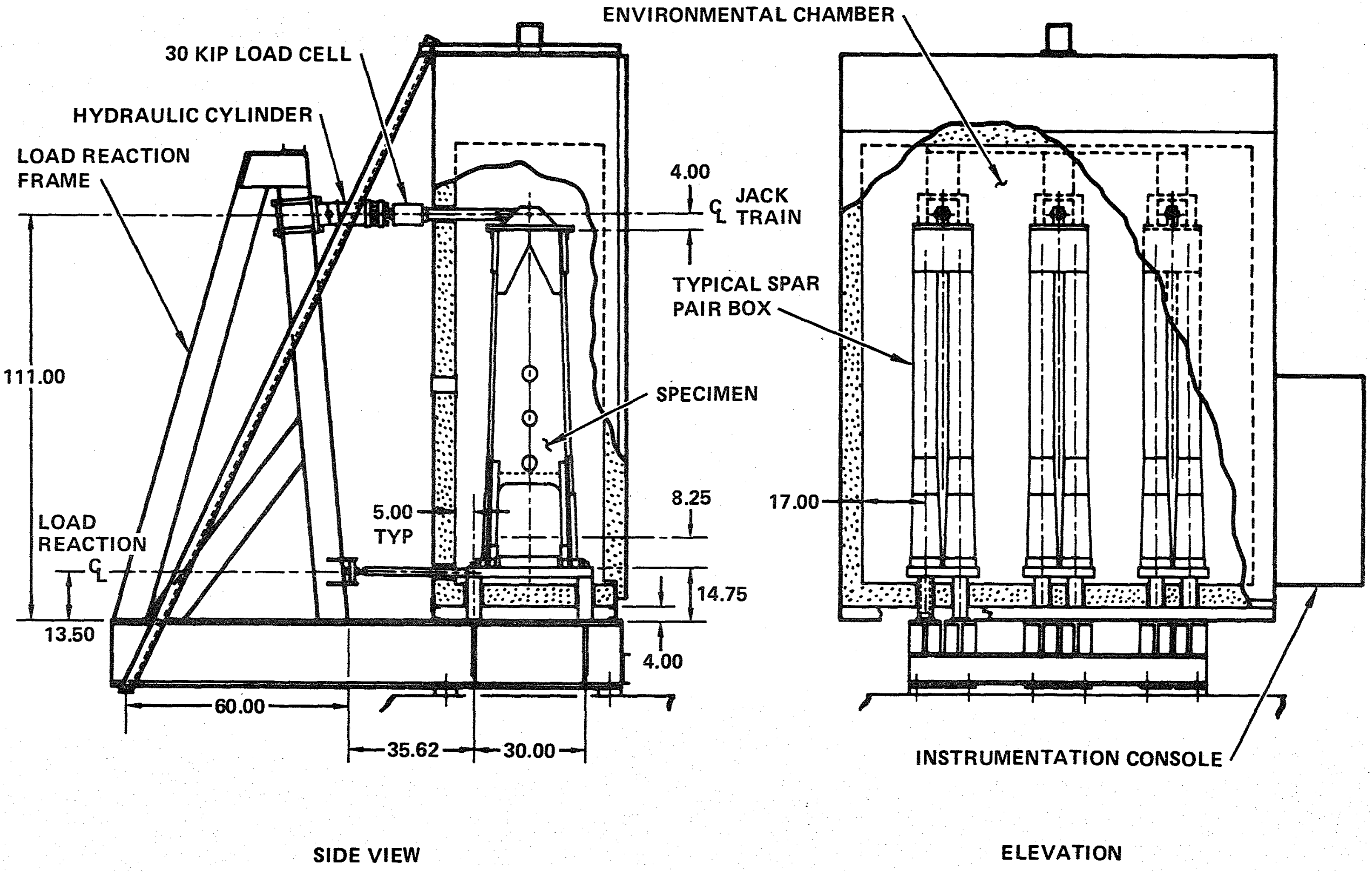


Figure 3-3. Test Setup - Spar Durability

A cover environmental chamber is shown mounted on its load reaction frame in Figure 3-4. Air flow in this chamber is from the right side ducts as opposed to vertically in the spar chambers. Details of the installation of the covers in the chamber are shown on Figure 3-5. The specimens are shown mounted in back-to-back pairs to enclose the surfaces. This is to simulate conditions in the actual fin, keeping the airflow on the outer surfaces of the panels and a more stagnant condition in the enclosed area. Although these specimens are mounted in pairs they are not connected from a load transfer standpoint. Each has its own separate loading jack and failure of one would not affect the other.

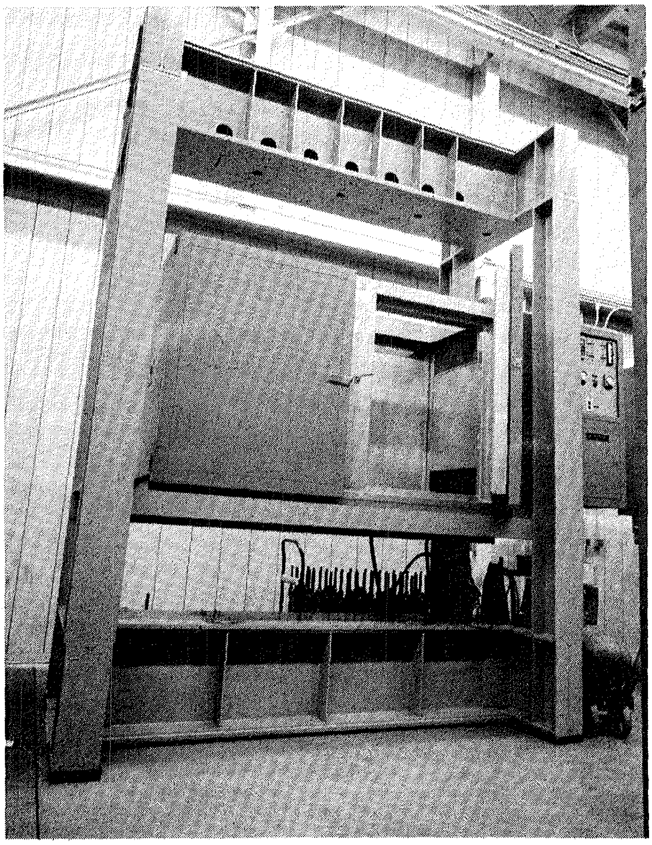


Figure 3-4. Cover Test Chamber Mounted in Load Reaction Frame

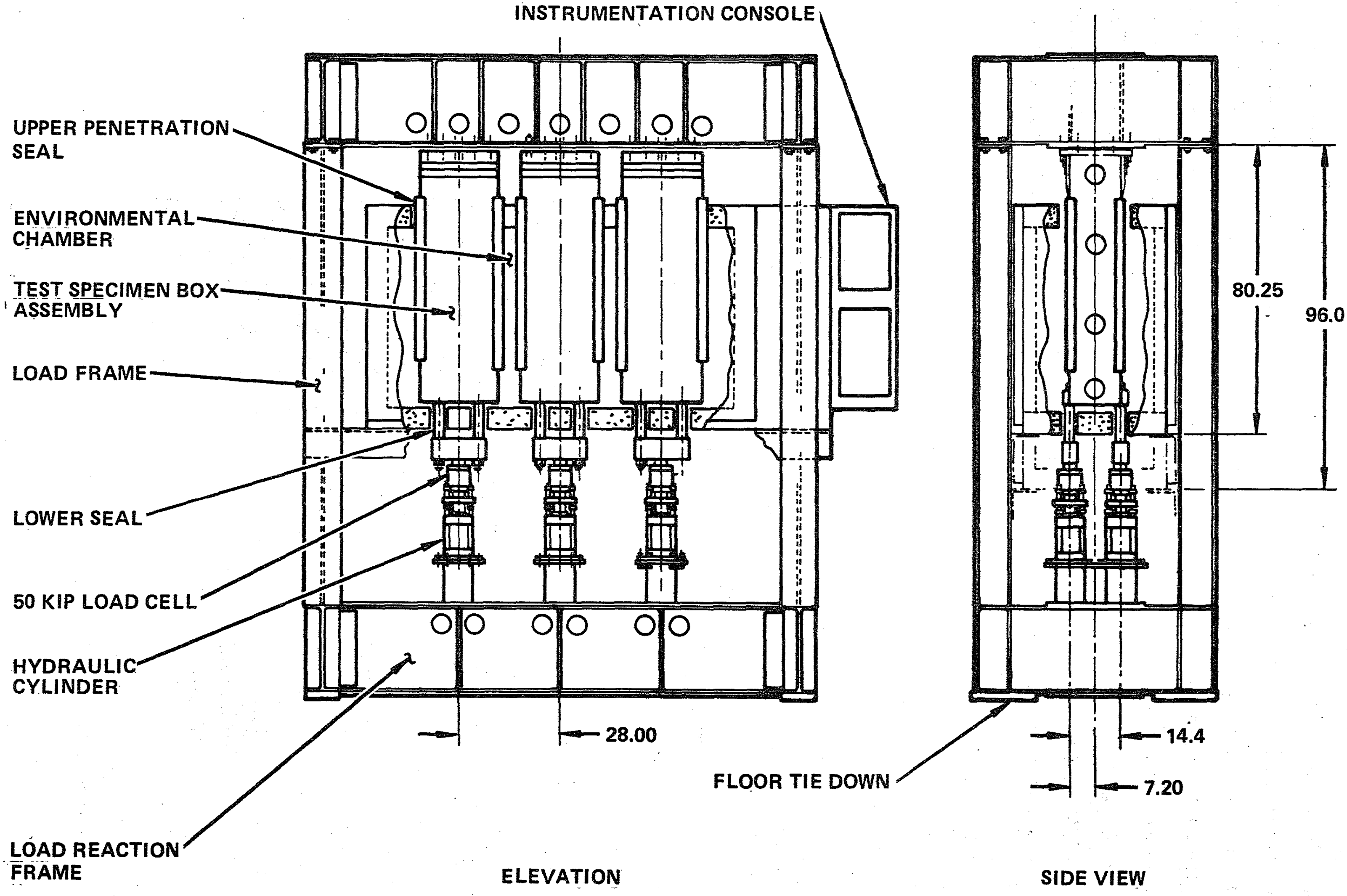


Figure 3-5. Test Setup - Cover Durability

3.2 PRVT COVER COMPONENTS

3.2.1 Cover Fabrication

Fabrication of the twenty hat stiffened panels for PRVT test began immediately following successful completion of the H25 static test. By year end, eight of the twenty cover PRVT specimens had been fabricated. The remaining twelve specimens are scheduled for completion during the next reporting period. Two PRVT cover specimens are shown in Figure 3-6.

3.2.2 Cover Test Preparation

Details of the assembly of the covers for the durability test are shown in Figure 3-7. The assembly of the cover specimen for durability testing will begin during the next reporting quarter. The test fixtures used in testing H-25 (para. 2.4.1) are being refurbished for use in the static test portion of the PRVT program.

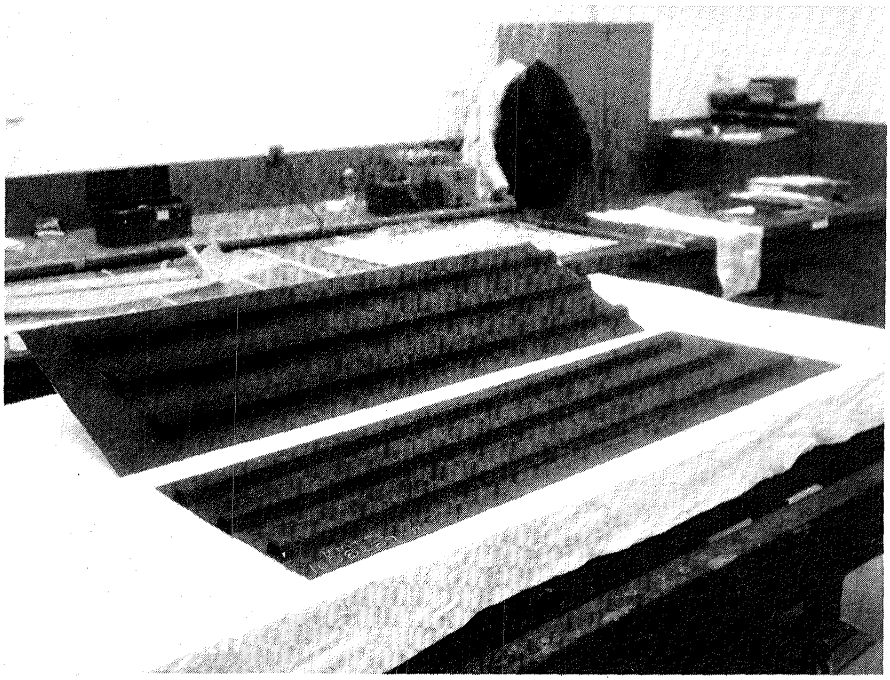


Figure 3-6. Cover PRVT Specimens

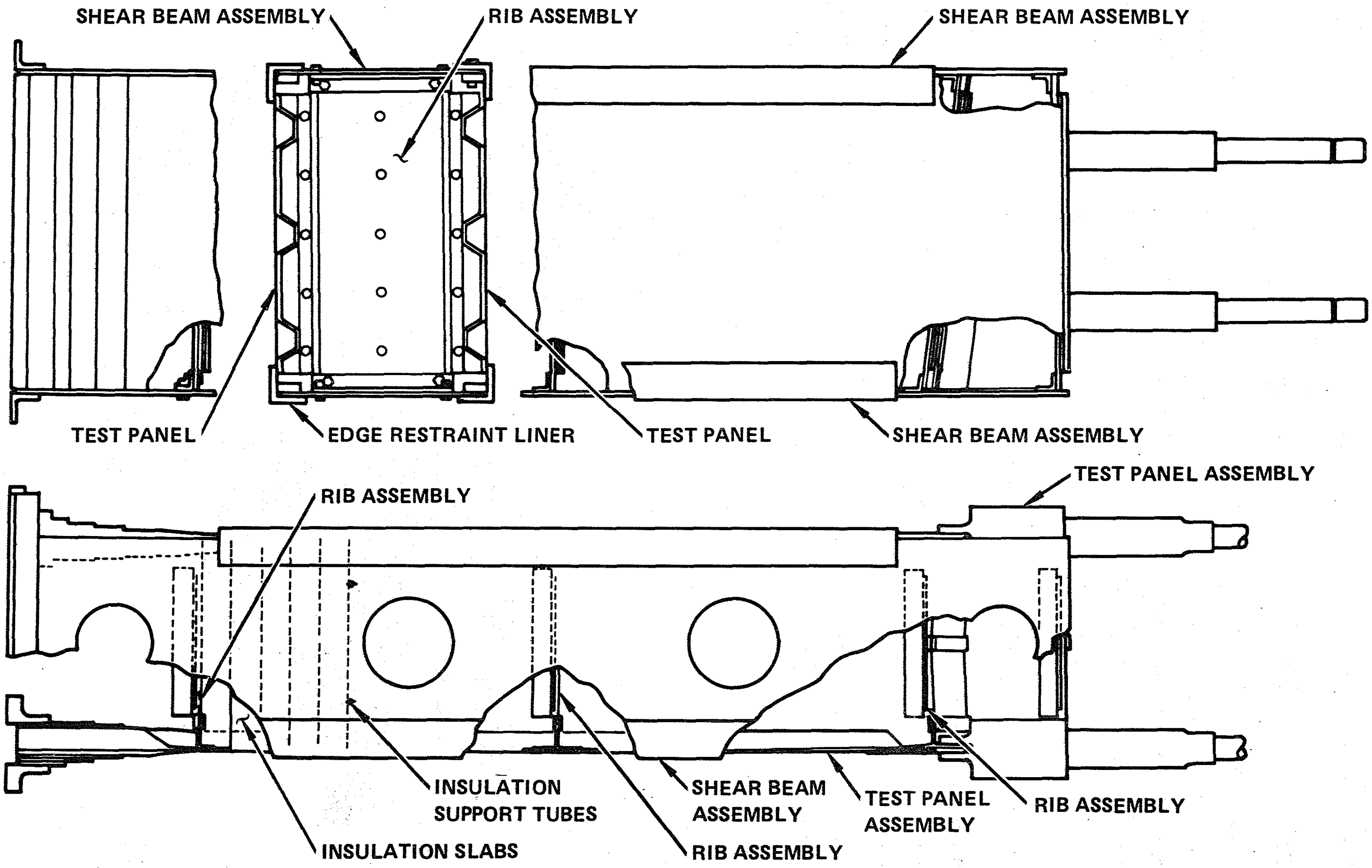


Figure 3-7. Cover Durability Test Specimen Assembly

3.3 PRVT SPAR COMPONENTS

3.3.1 Spar Fabrication

Sixteen PRVT spar specimens have been fabricated and are in preparation for test. The remaining four spars will be completed for shipment from the Lockheed-Georgia company early next year.

A good data base is being accumulated from the process control specimens cut out of the spar web access holes. Figure 3-8 summarizes the compression, short beam shear, resin content and thickness test results. Compression tests of spar numbers 6 and up were made with the modified FED STD 406 test fixture shown in Figure 3-9. As seen in Figure 3-8, the short beam shear appears to be sensitive to resin content.

The edge of the third access hole was delaminated on spar number 6 when the disc was being cut out. A repair was made using graphite cloth and 5208 resin. The area around the hole was bagged, sealed, and cured in an oven under vacuum pressure. The repair is shown in Figure 3-10.

3.3.2 Spar Test Preparation

Details of the assembly of the spars for durability testing is shown in Figure 3-11. Assembly of the spars for durability testing is in progress. Figure 3-12 and 3-13 show the spar subassembly. This subassembly is installed in the fixture shown in Figure 3-14 and 3-15 for assembly into spar pairs required for durability testing.

During shipment of the first lot of PRVT spar specimens, two were damaged. A contact pulse-echo technique was used to determine the extent of damage to the spars. This technique was used to simulate field conditions. Spar specimen number 2 (durability test specimen), Figure 3-16, showed evidence of delaminations on two stiffeners. The delaminations were between the stiffener and web. Spar specimen No. 3 (static test specimen), Figure 3-17, showed evidence of delaminations on one stiffener. This delamination was nearly continuous across the length of the stiffener-to-web interface. Additionally, the vertical leg showed evidence of extensive delamination.

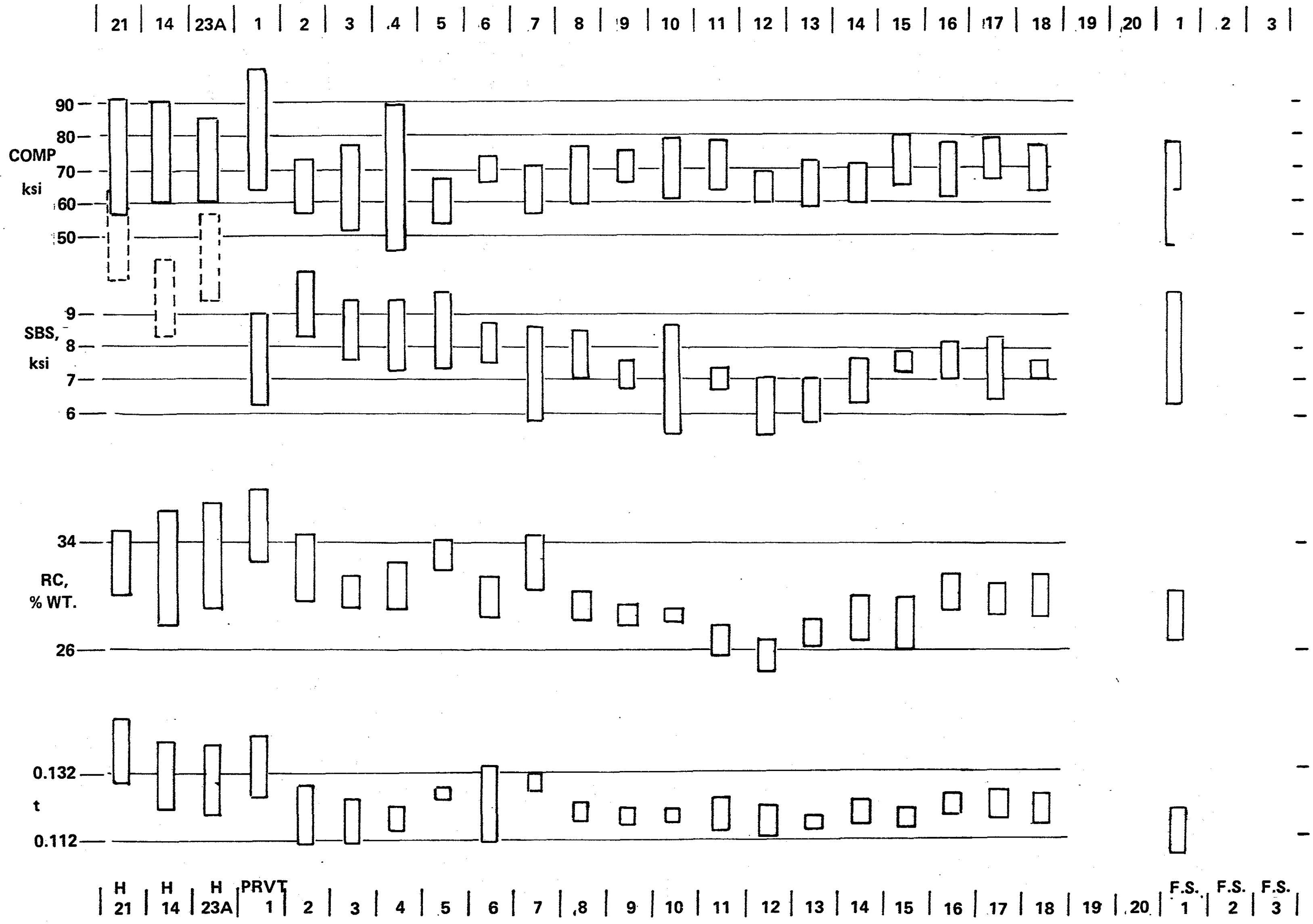
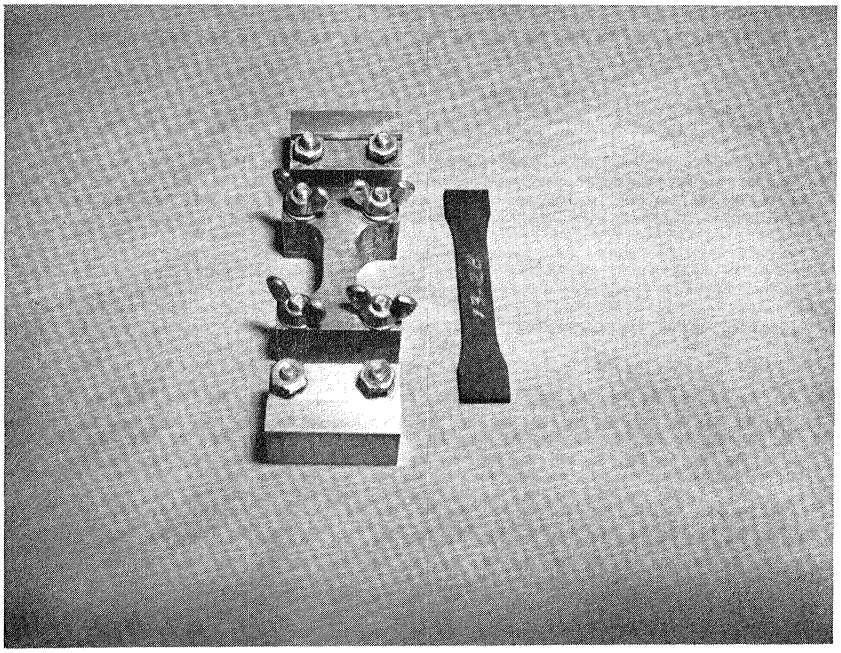
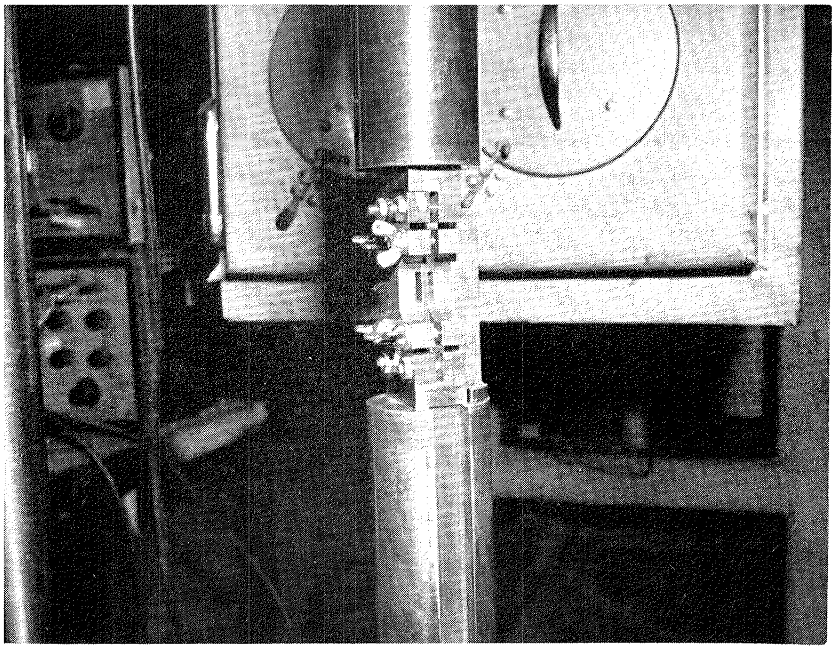


Figure 3-8. Summary of Process Control Data from Spar Web Access Holes

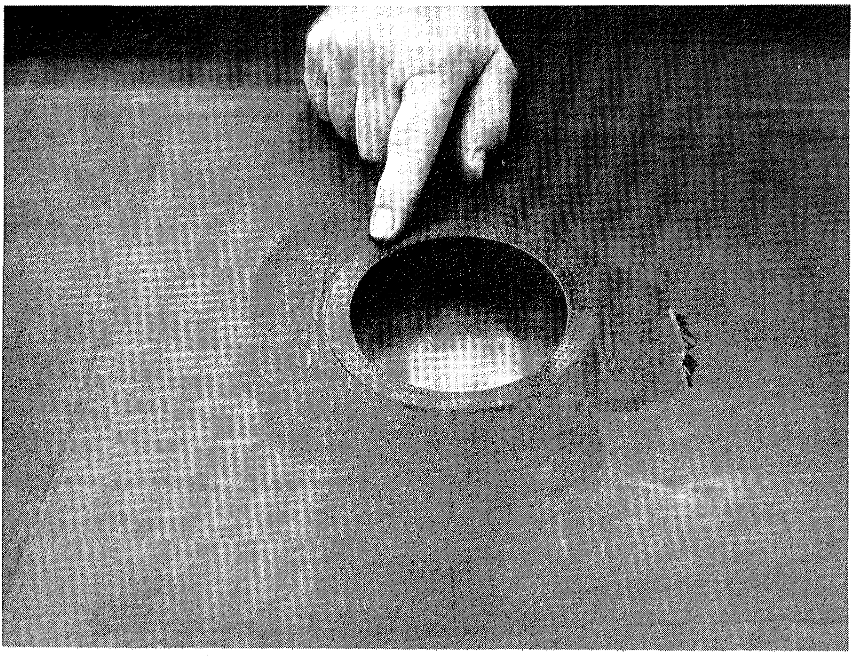


RL 4147-1



RL4147-5

Figure 3-9. Modified FED STD 406 Compression Text Fixture



RL 3991-3

Figure 3-10. Repaired Edge of Access Hole in PRVT No. 6

A stiffener repair procedure has been developed and used on the two PRVT spar specimens. These repairs will be evaluated as part of the PRVT static and durability test program.

3.4 REPAIR PROCEDURE FOR DELAMINATED STIFFENERS

The extent and location of damage to the stiffener and surrounding area is determined by visual and NDI techniques. If damage results from delamination within the laminate forming the outstanding leg or from general web-to-stiffener disbond or delamination, the stiffener may be repaired by using the following repair procedure:

1. The surface to be bonded must be free of any debris, grease, oil or other foreign material.
2. Smooth all rough exterior surfaces which will be in contact with aluminum repair doublers.
3. Fabricate aluminum doublers to the configuration and quantity shown in Figures 3-18 and 3-19.

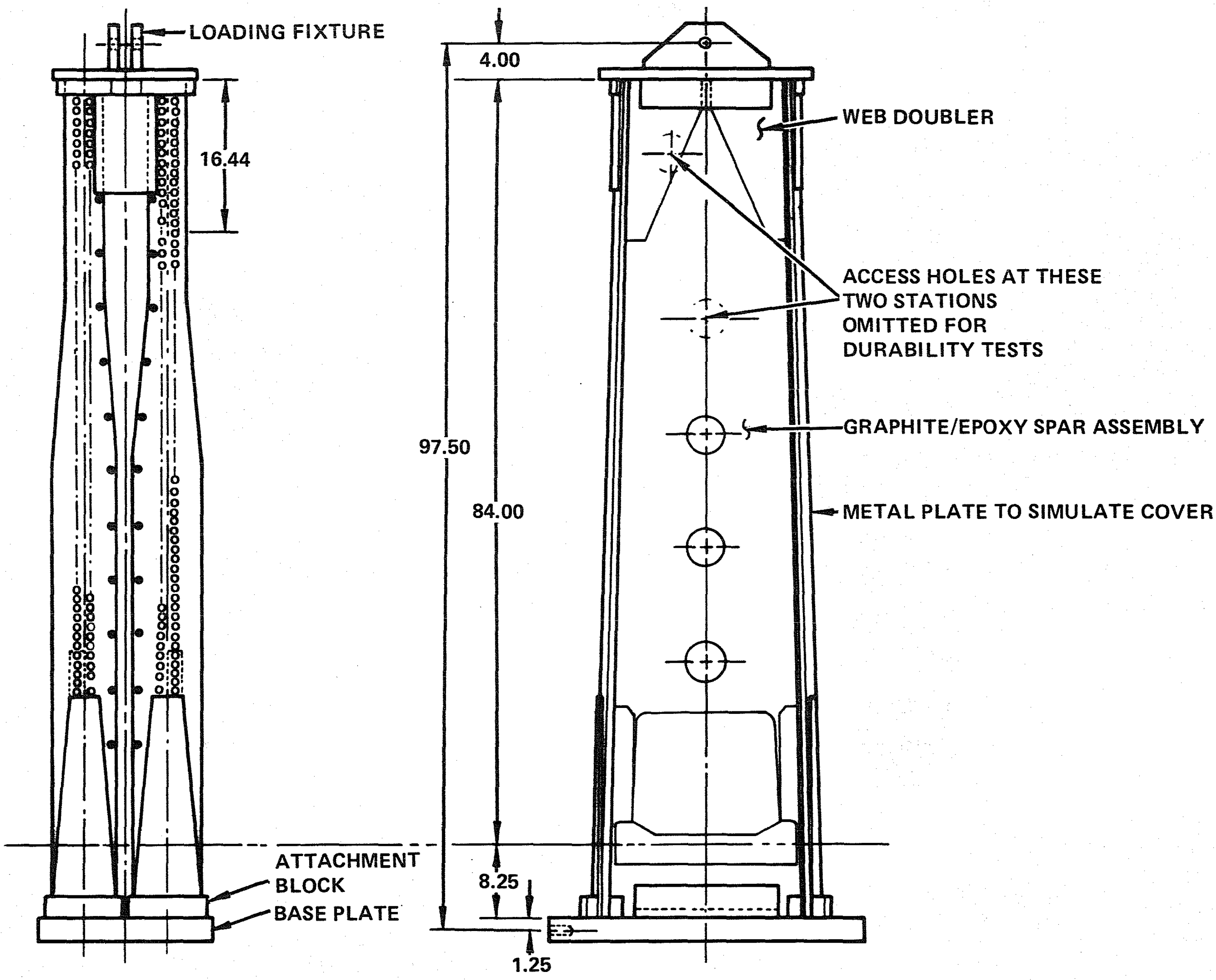


Figure 3-11. Spar Durability Test Specimen Assembly

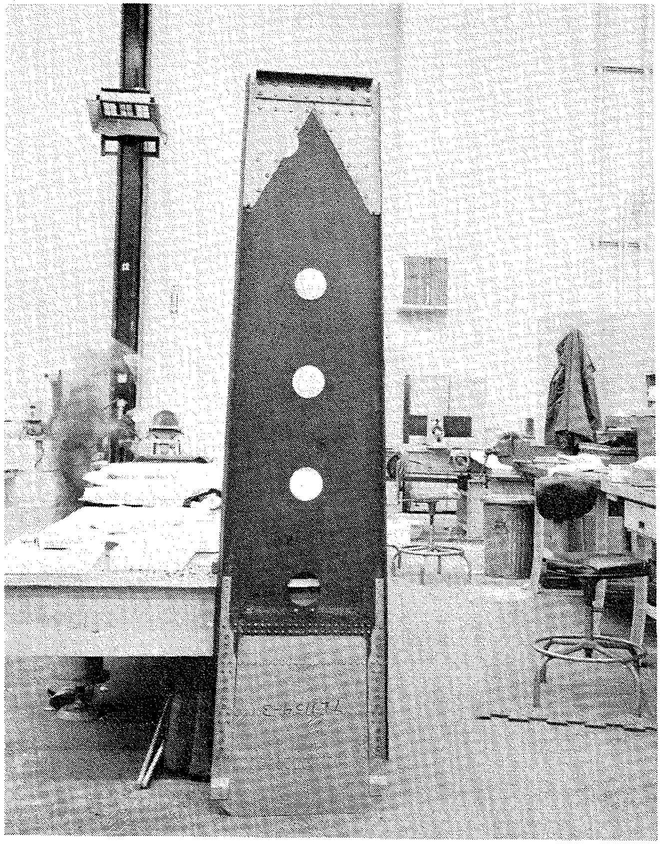


Figure 3-12. Front Side of Spar with End Attachments

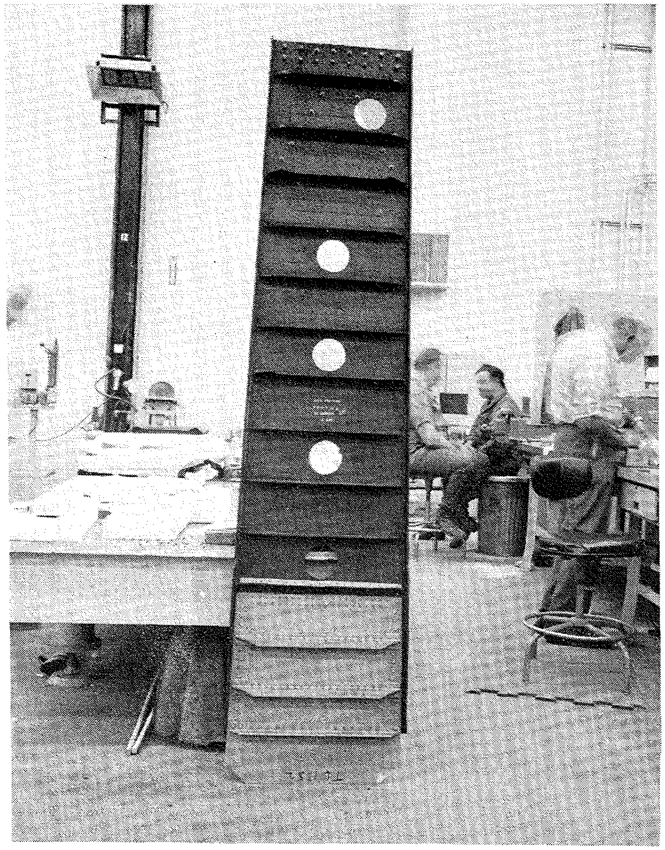


Figure 3-13. Back Side of Spar in Figure 4.

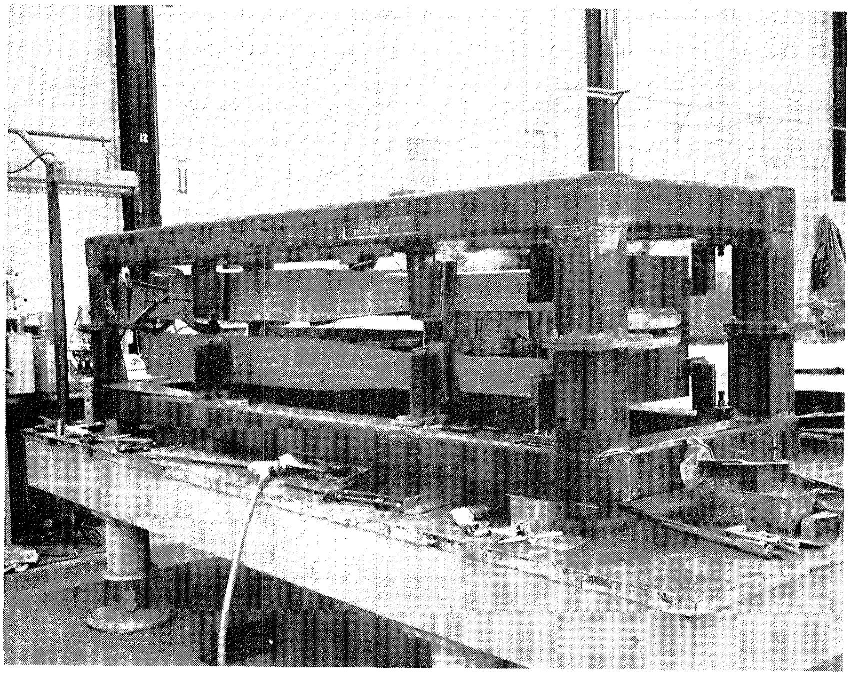


Figure 3-14. Spar Pair Assembly Fixture

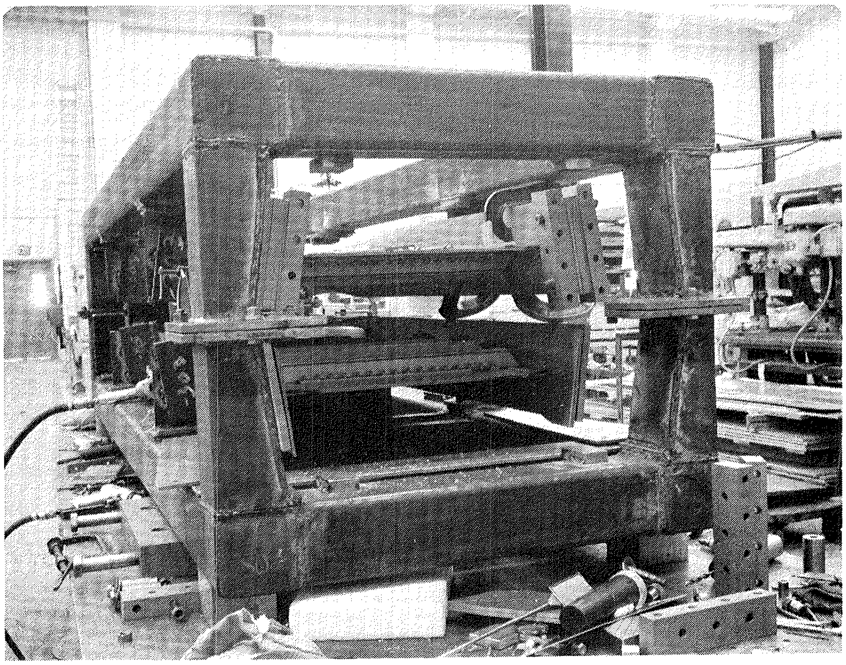


Figure 3-15. Spar Assembly Fixture Viewed from the Root End

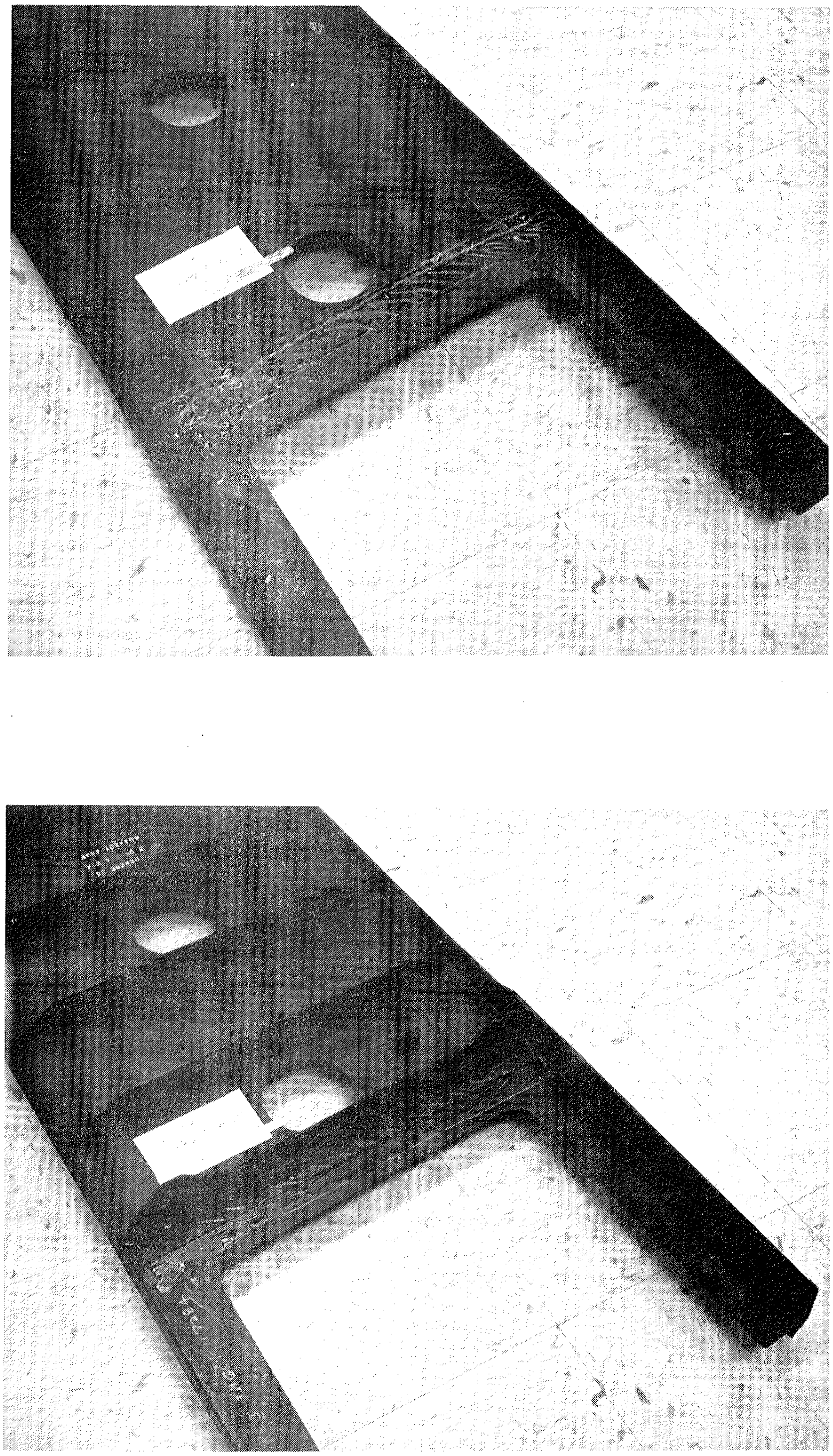


Figure 3-16. PRVT Spar Specimen No. 2, Ultrasonic Inspection Results - Crosshatched Areas Indicate Delamination

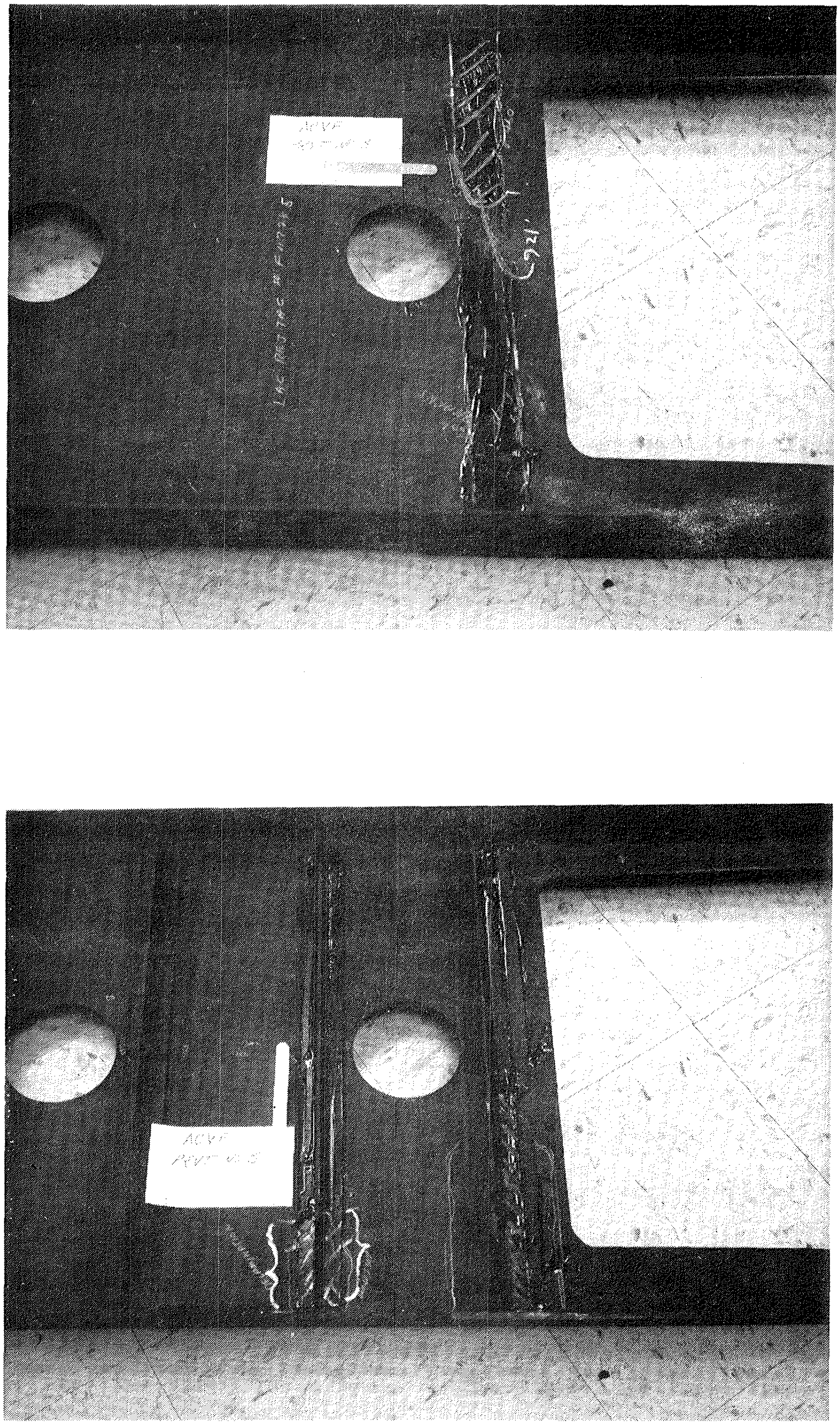
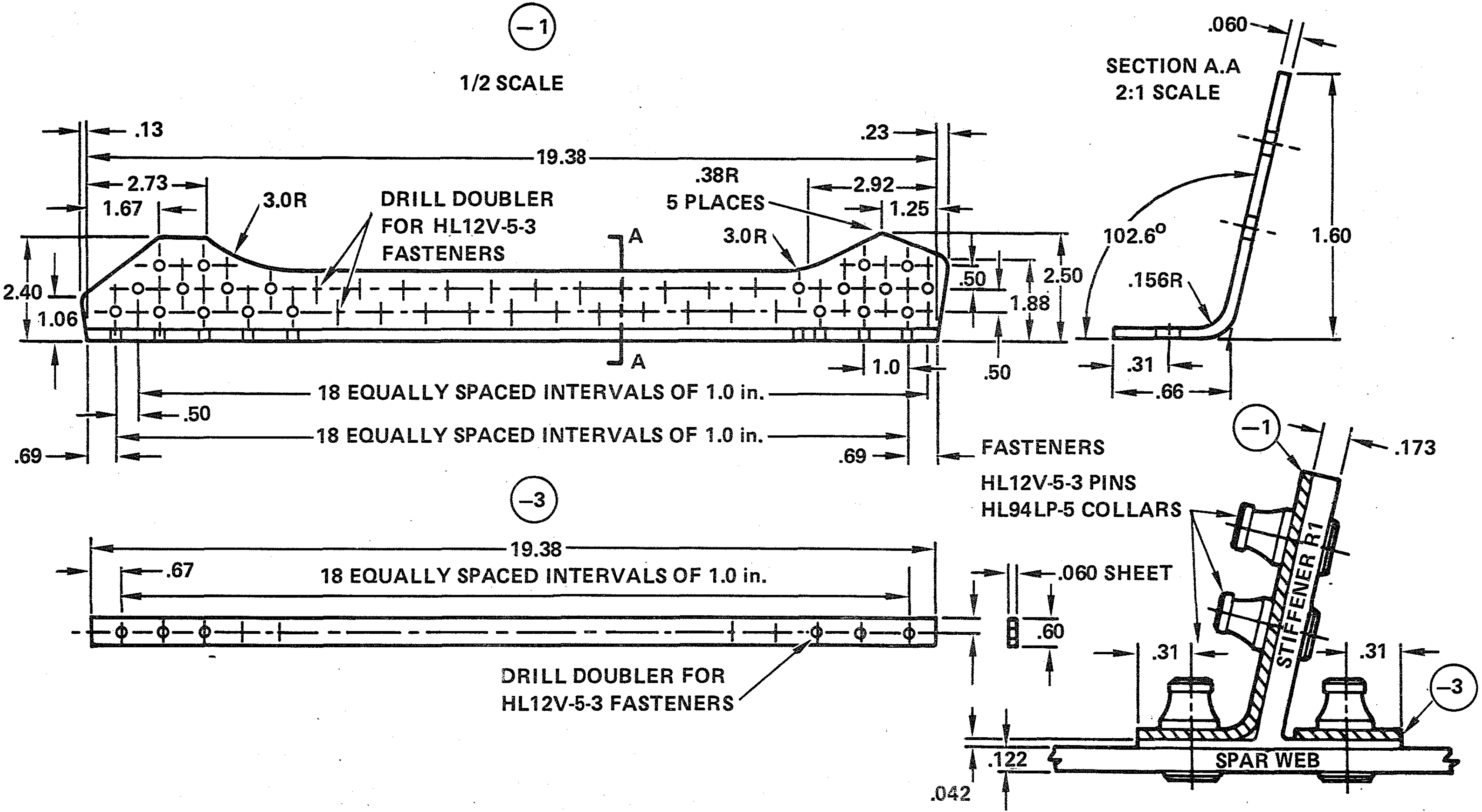


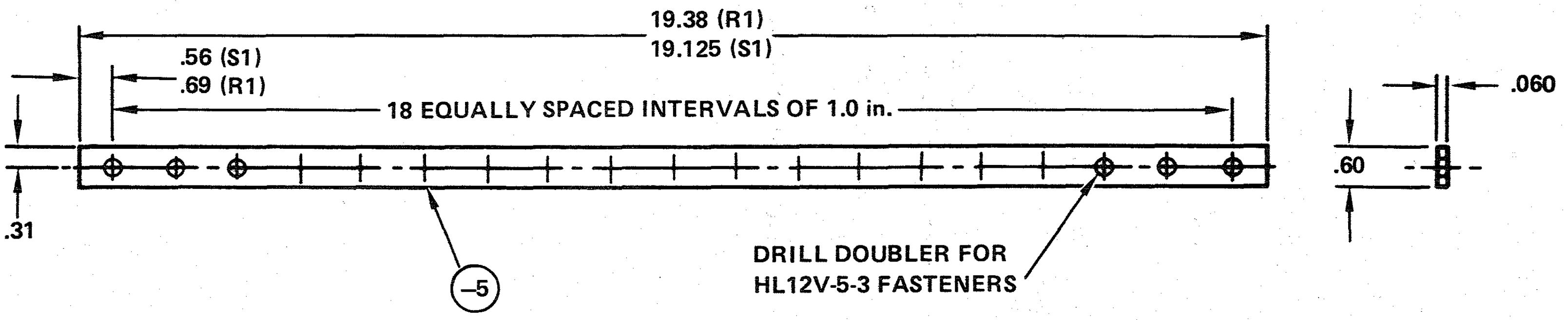
Figure 3-17. PRVT Spar Specimen No. 3, Ultrasonic Inspection Results - Crosshatched Areas Indicate Delaminations



NOTES

1. MATERIAL −3 2024-T3, −1 2024-T4 (H.T. AFTER FORMING BOND RADIUS)
2. FINISH CODE AA-83
3. ALL FASTENERS TO BE INSTALLED WET

Figure 3-18. Detail of Aluminum Doublers and Installation on PRVT Spar No. 2



NOTES:

1. MATERIAL 2024-T3
2. FINISH CODE AA83
3. ALL FASTENERS TO BE INSTALLED WET

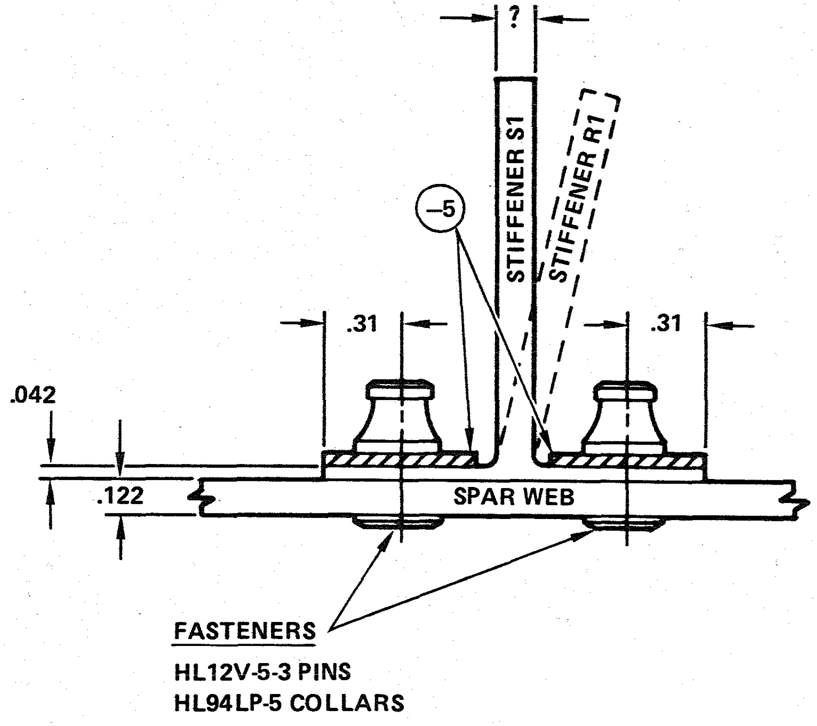


Figure 3-19. Detail of Aluminum Doublers and Installation on PRVT Spar No. 3

4. If the delaminated surface is accessible, wipe clean with a lint free cloth soaked in MEK.

5. Mix Hysol 9309 (250°F curing) adhesive per directions on the container. (23.1 ratio)

6. If the delaminated surface is accessible, apply adhesive with a spatula applicator, if not, inject the adhesive directly into the damaged area with a hypodermic syringe.

7. Assemble metal and graphite parts, drill holes to final size.

8. Disassemble and clean all parts with MEK.

9. Apply adhesive to both graphite and aluminum mating surfaces. Assemble. Insert undersized fasteners in selected holes for clamp up during curing.

10. If circumstances permit, bag and apply a vacuum to the damaged zone so as to improve the diffusion of the adhesive throughout the joint.

11. Heat area to 250°F for 15 minutes or 180°F for 1 hour. (A heat lamp or other portable device is acceptable.)

12. Upon cooling, remove undersized fasteners and install wet HL-12V fasteners in all holes.

SECTION 4

PHASE IV - MANUFACTURING DEVELOPMENT

With test specimen fabrication nearing completion, manufacturing emphasis at both Lockheed-California and Lockheed-Georgia companies is directed to fabrication requirements for full-scale components. At Lockheed-California the full-scale cover layup tools completed autoclave thermal profile checks. At Lockheed-Georgia, the first full-scale spar was molded.

4.1 COVERS

The 26 x 10 foot layup tools (MBFs) were run through a cure cycle simulating the various temperature plateaus, heat rise rates, and pressure conditions that will be encountered in the cure cycle. Thermocouples were attached to the top and bottom surface of each MBF to provide a thermal profile of the tools during all phases of the cure cycle. The tools were vacuum bagged and run in the large autoclave. When the tool temperatures reached 250° ± 10°F, 85 psi autoclave pressure was applied. After 5 minutes at 85 psi, plus 24" Hg, and 250° ±10°F, a leak check was made.

Analysis of the thermocouple and pressure charts revealed the following conditions:

- The autoclave has more than adequate capability to meet the heat-up rates, temperature limits, and pressure requirements of the cure cycle.
- The temperature differential between the top and bottom surfaces of the tools was consistent at each thermocouple location and ranged between 5° min. to 10°F max. during heat rise from 100°F to 210°F and again during heat rise from 210°F to 250° ±10°F. However, during the dwell period at 250° ±10°F the temperature differential narrowed down to 5°F max. During heat rise from 250°F to 350° ±10°F the temperature differential was 5°F max. With this information it is now possible to use the thermocouples which are semi-permanently attached to the bottom surface of the tools for the purpose of monitoring specified cure parameters during cure of the first Tool Try Part.

- The autoclave is large enough to simultaneously cure two full-size hat/cover assemblies in one autoclave load.
- The pressure leak check was considered acceptable since there was no evidence of leaks with respect to the vacuum bag, bag sealant, and the basic tools. The inflatable bladders intended for use on the Tool Try were not checked at this time but will be independently pressure checked.

Figure 4-1 shows the full-size MBFs, vacuum bag, and thermocouples used during the thermal profile and heat/pressure check. The autoclave instrument panel is in the right background. Figure 4-2 shows the relative size of the MBFs and the large autoclave.

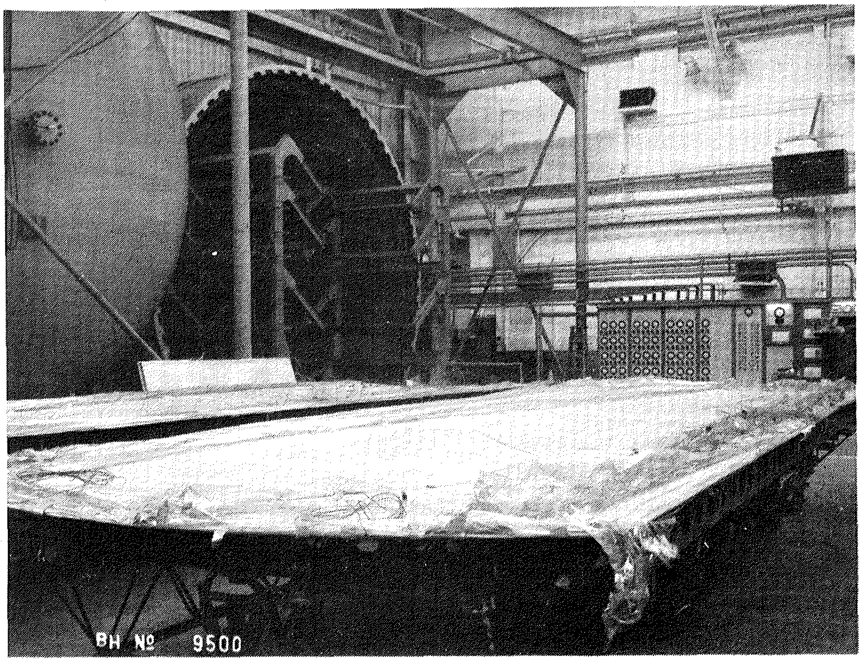


Figure 4-1. Full-Scale MBFs

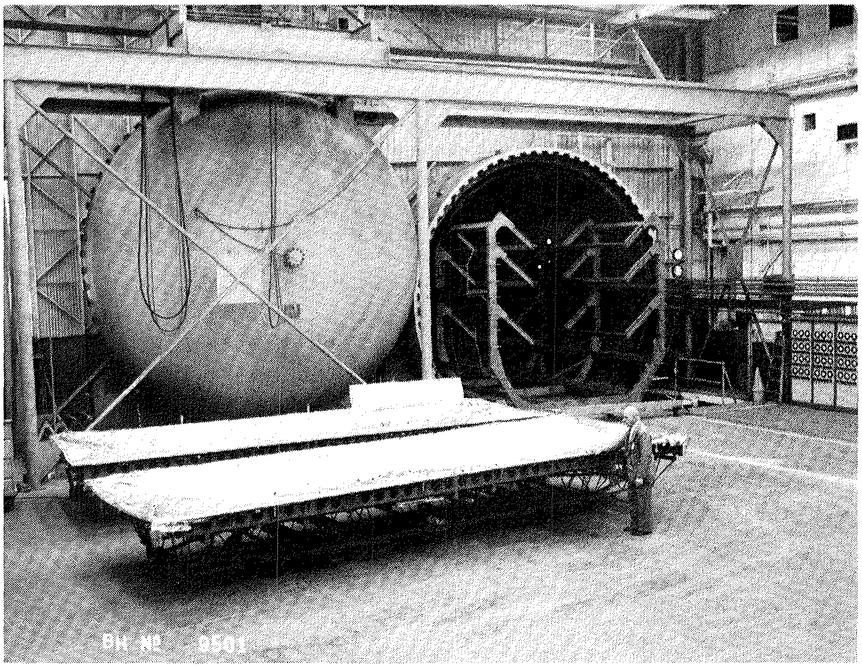


Figure 4-2. Full-Scale MBFs by Large Autoclave

4.2 SPARS

4.2.1 Front Spar

Fabrication of the first full-size front spar was accomplished in this quarter. Prior to fabrication of the first graphite spar, a stiffened web fiberglass part was made in the tool for preliminary evaluation, see Figure 4-3.

The sequence of fabrication of the first graphite spar is shown in Figures 4-4 through 4-17. Figure 4-4 shows the preplied stiffeners being loaded into the tool. Figure 4-5 shows the spar web laid on the stiffeners. The armalon breather is then laid on top of the web and the island blocks and rubber placed on top of the web. After web and stiffeners are secured the spar caps which had been preplied in the cap rails are attached to the web as shown in Figure 4-6. Final hard compaction of all the internal parts is shown in Figure 4-7. Figure 4-8 shows the cover placed on the tool and Figure 4-9 shows the tool being prepared for bagging.

After curing, the tool is disassembled. Figures 4-10 and 4-11 show the lower and upper ends of the spar inside the tool cover. The removal of the rubber blocks is shown in Figure 4-12 and removal of the lower steel island blocks is shown in Figure 4-13. Once all the internal tool parts have been removed from the stiffener side of the spar, the baseplate is put back and the tool is turned over. The cover is then lifted off and the spar is removed as shown in Figure 4-14.

The spar as removed from the tool is shown from the upper end in Figure 4-15 and from the lower end in Figure 4-16. The forward side is shown in Figure 4-17.

Discs are cut from the spar web to provide access holes in the spar web and to provide specimens for process control. The results of the process control tests performed on specimens cut from the discs are shown in Table 4-1. The completed spar is shown in Figure 4-18.

4.2.2 Rear Spar

The tooling for the rear spar is nearing completion. Figures 4-19, 4-20 and 4-21 show the fit-up of the rear spar tool prior to pouring the rubber. Figure 4-19 shows the island blocks placed on the base plate and inside the tool cover. The blocks on the base plate and the cast rubber form the stiffener side of the spar web. The blocks inside the cover are used to mold the smooth side of the web. Figure 4-20 shows a section of the dummy web being placed inside the tool cover. In Figure 4-21, the island blocks (shown on the base plate in the upper part of the figure) are fitted on top of the dummy web inside the tool cover. The spar cap rails within the dummy spar cap are also fitted inside the tool cover as shown in the lower half of Figure 4-21. These parts are then secured with "C" clamps prior to pouring the rubber.

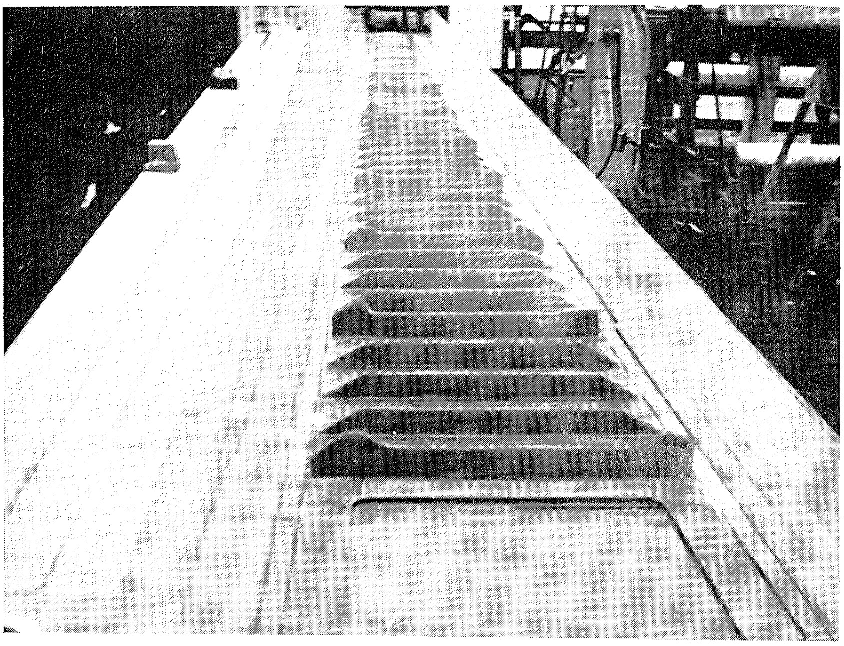
TABLE 4-1. SUMMARY OF FRONT SPAR PROCESS CONTROL DISCS

Disc No.	Location VSS	SBS (6.0 ksi)*	Flex 52 ksi △1 40 ksi △2 30 ksi △3	Comp 50 ksi △1 42 ksi △2 33 ksi △3	RC (26-34%)*	SG (1.54 - 1.60)*
1	100	7.9	72.6	65.3	26.7	1.60
		8.6		61.8	26.8	1.60
2	124	7.3	77.2	76.6	30.4	1.57
		7.9		68.6	30.3	1.57
3	155	7.5	75.3	72.5	27.9	1.58
		8.0		76.6	27.6	1.58
4	175	7.3	74.9	46.6 79.2 △4	29.8	1.57
		7.3		68.9	29.4	1.57
5	200	6.3	76.6	76.6	27.3	1.59
		8.3		73.4	26.7	1.59
6	226	8.1	74.2	77.0	27.3	1.59
		9.7		73.5	28.3	1.58
7	252	8.7	58.1	69.8	27.5	1.59
		8.1		58.7	28.5	1.58
8	277	8.5	54.1	71.0	32.2	1.56
		8.9		64.6	32.8	1.56
9	303	7.2	39.8	49.4	33.3	1.56
		8.6		45.4	31.3	1.56
10	327	8.8	30.8	50.3	29.2	1.57
		6.8		49.5	30.5	1.57

*Acceptable values shown in Final Draft of PB80-580.

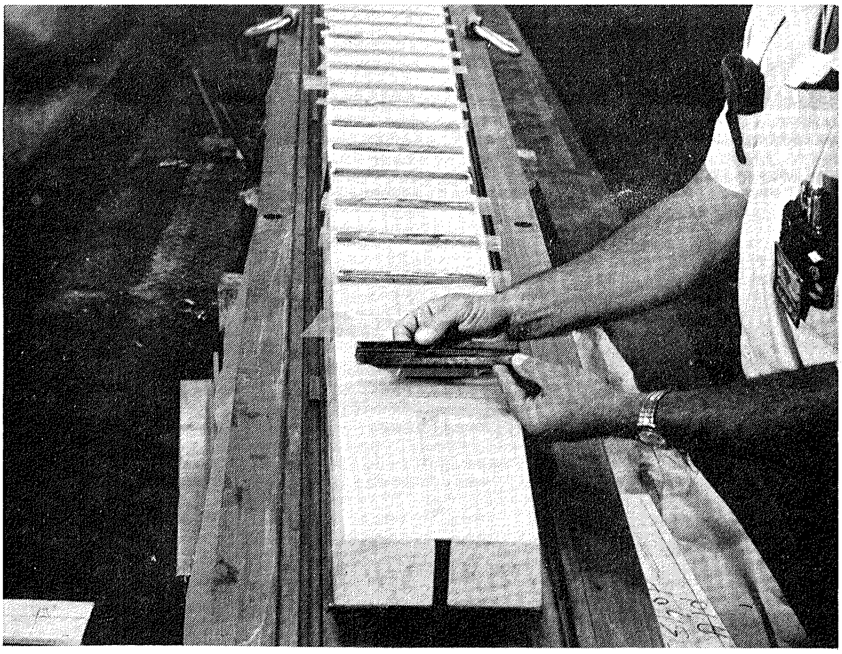
△1 Disc 1-6 △2 Disc 7, 8 △3 Disc 9, 10

△4 Adjacent Flex Specimen Tested in Compression



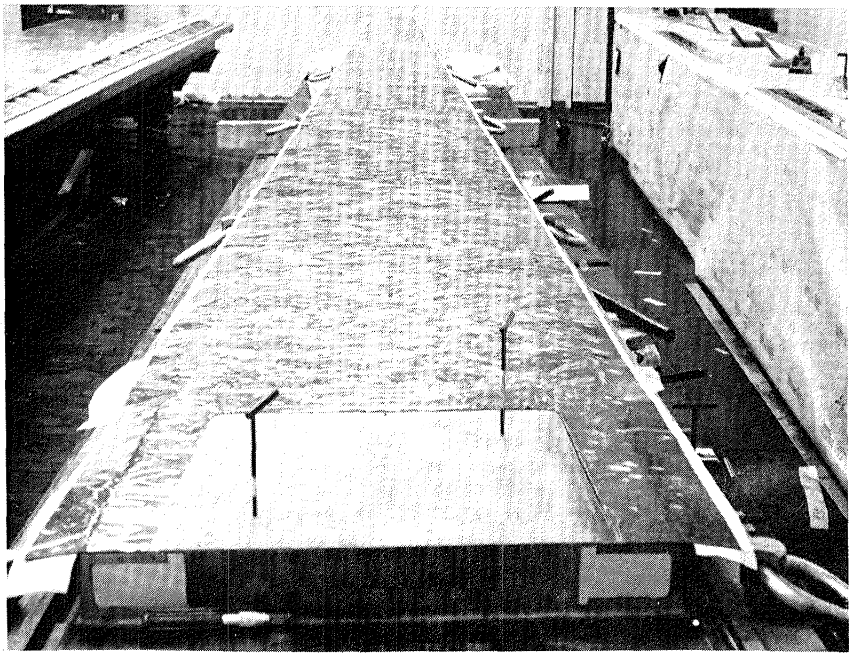
RL 3917-6

Figure 4-3. Fiberglass Tool Try Part



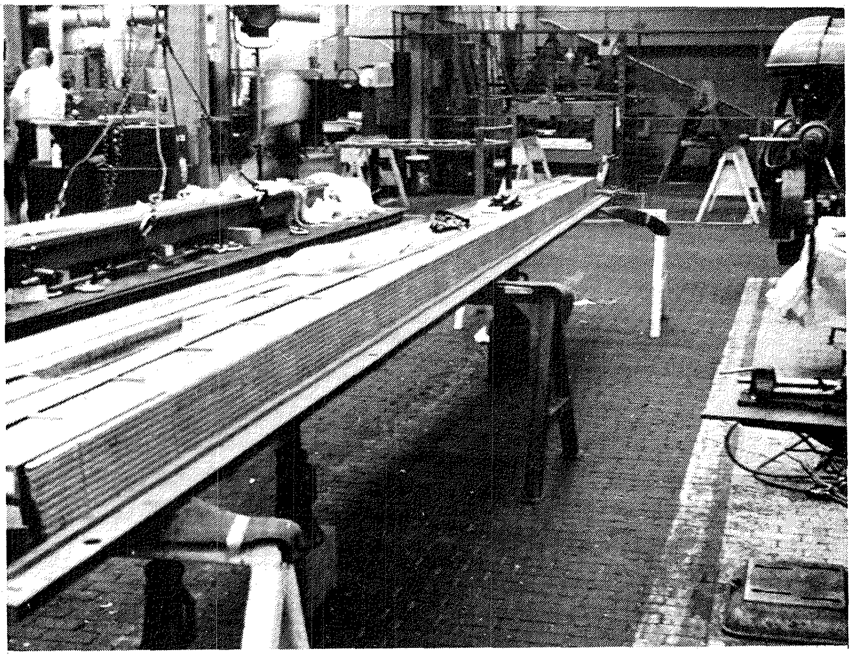
RL 3917-1

Figure 4-4. Loading Stiffeners in Front Spar Tool



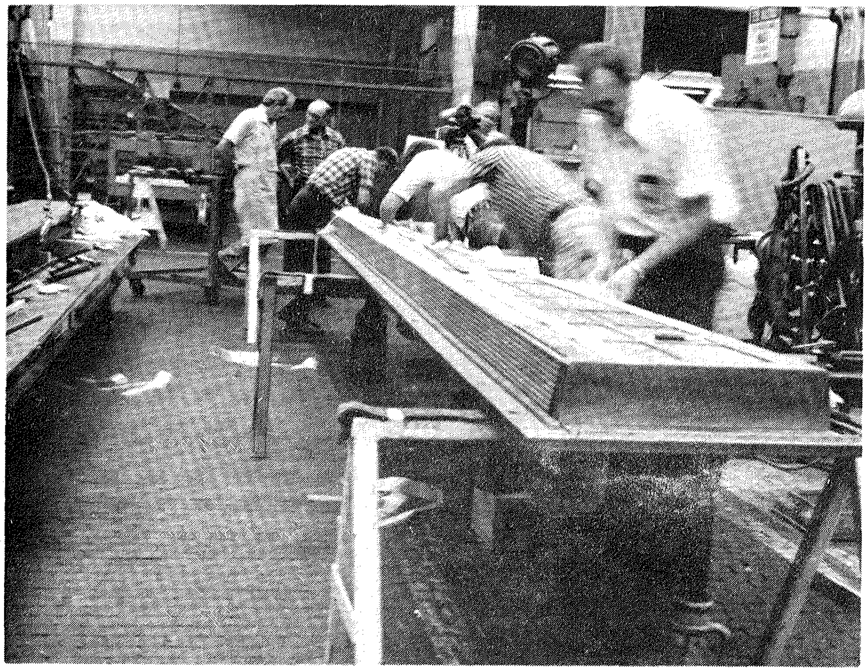
RL 3917-5

Figure 4-5. Spar Web Laid on Stiffeners



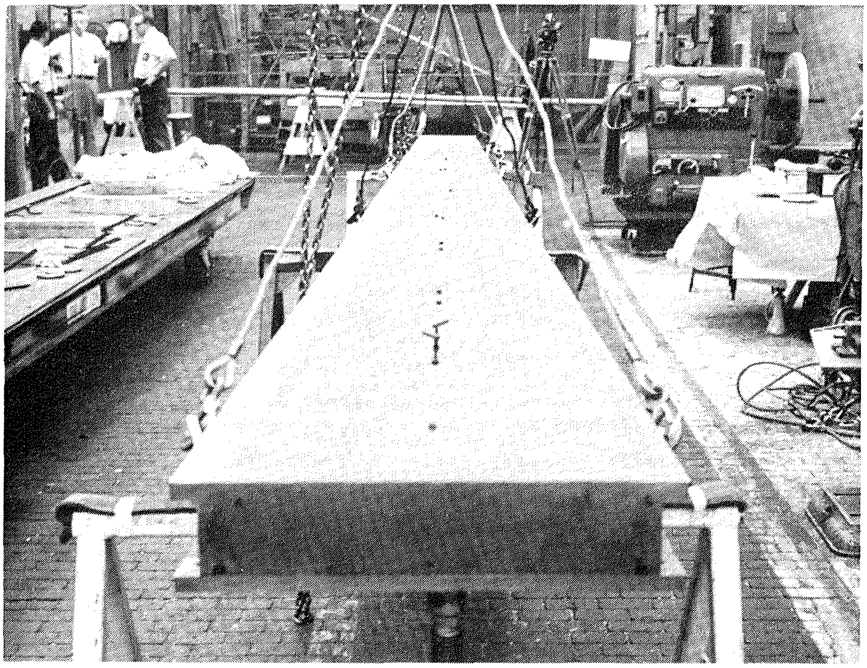
RL 3917-12

Figure 4-6. Spar Cap Rails Shown in Place



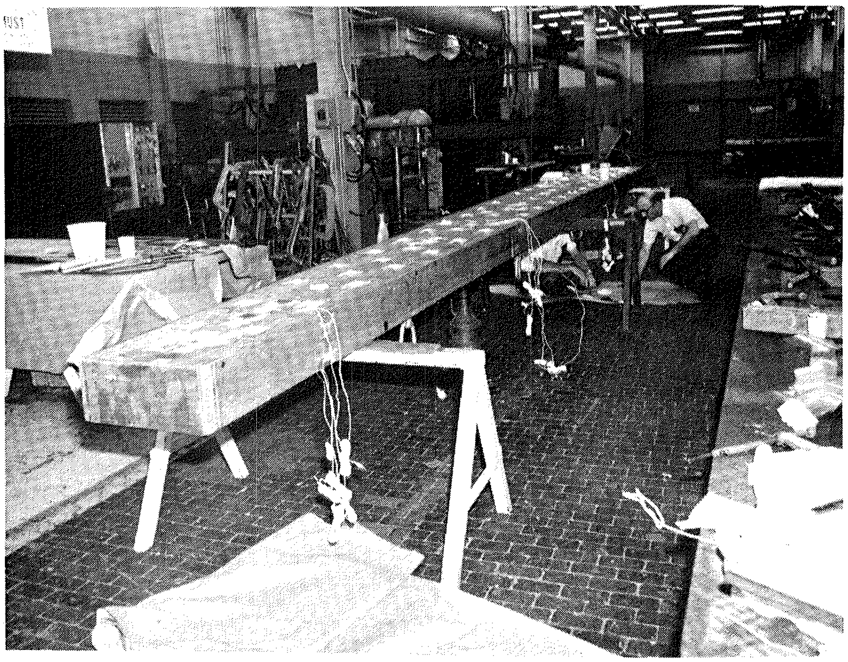
RL 3917-10

Figure 4-7. Hand Compaction of All Internal Parts



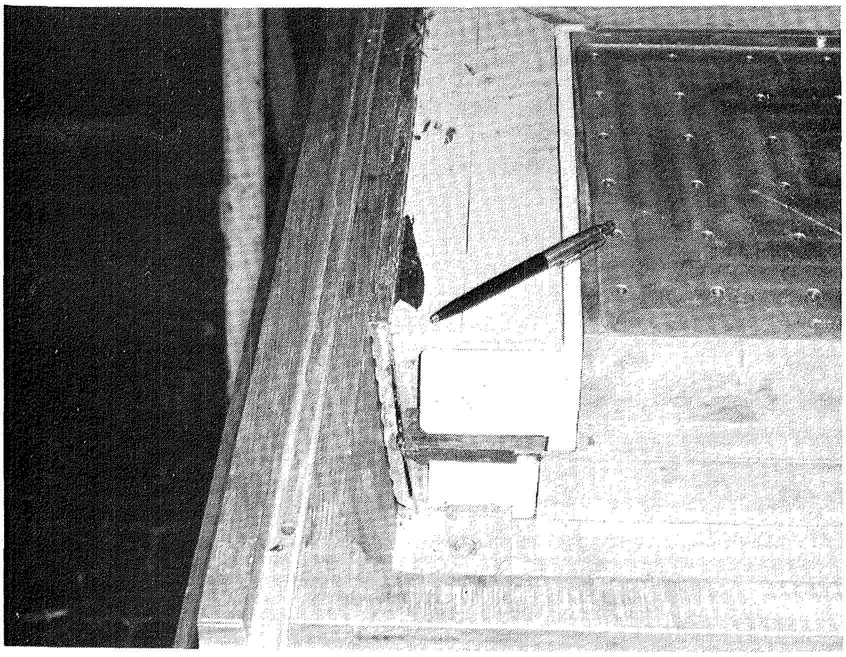
RL 3917-8

Figure 4-8. Cover Placed on Tool



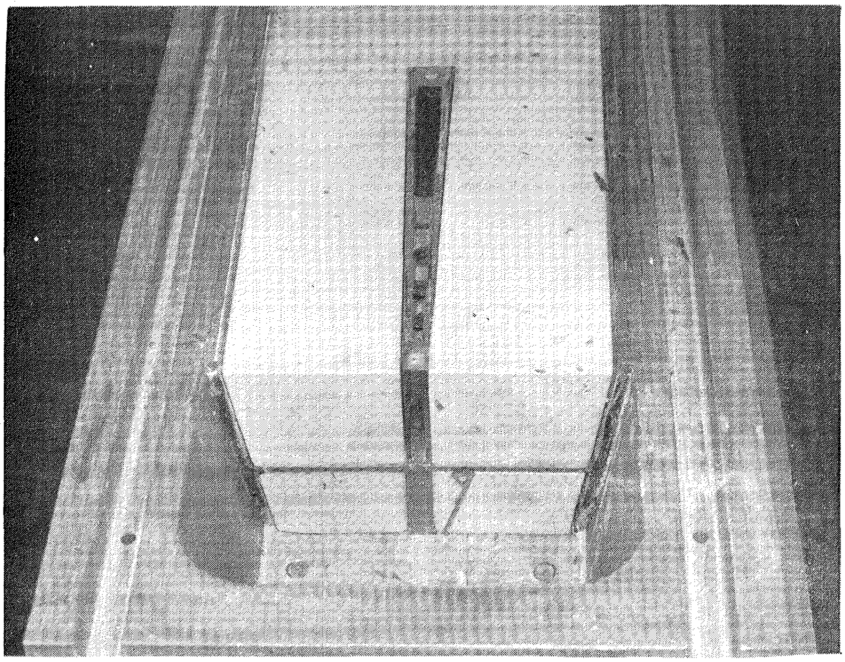
RL 3935-2

Figure 4-9. Prepared for Bagging



RL 3938-4

Figure 4-10. Spar Lower End Exposed Inside Tool

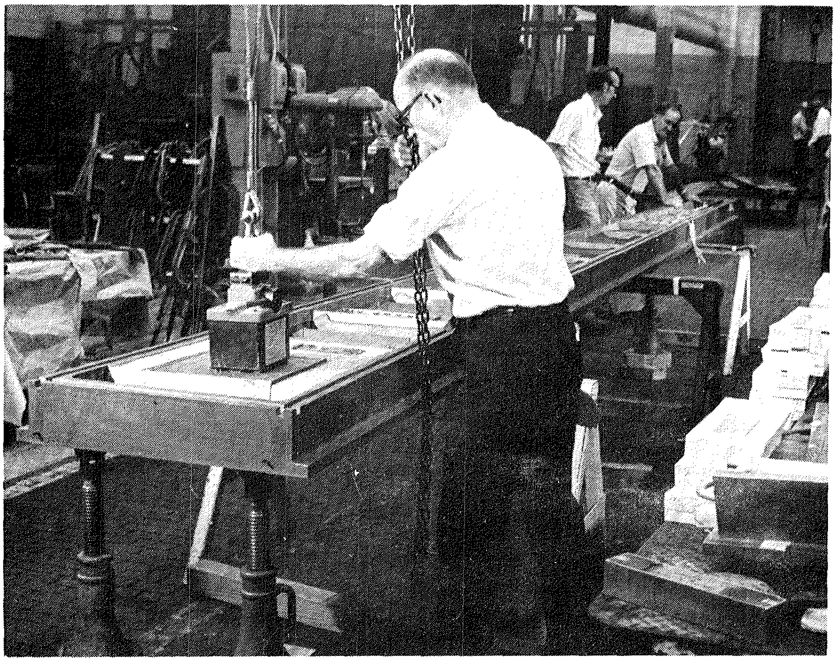


RL 3938- 3

Figure 4-11. Spar Upper End Exposed Inside Tool

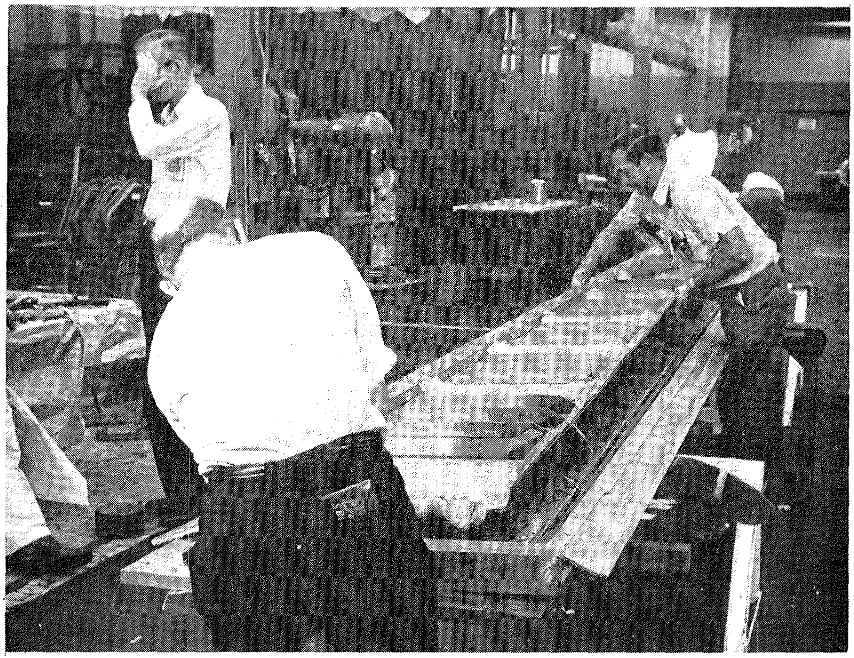
RL3938-5

Figure 4-12. Removal of Rubber Blocks



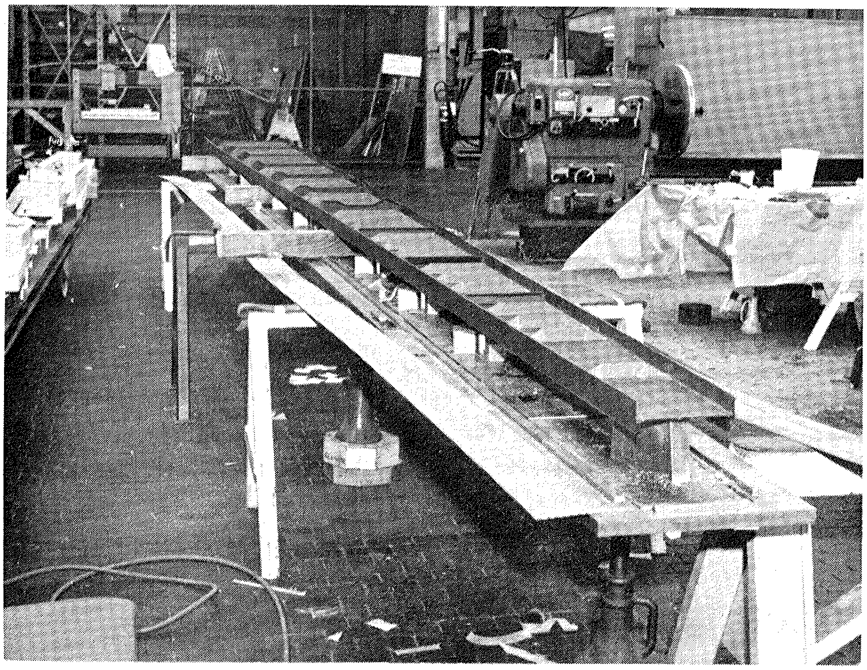
RL3938 - 14

Figure 4-13. Removal of Lower Steel Block



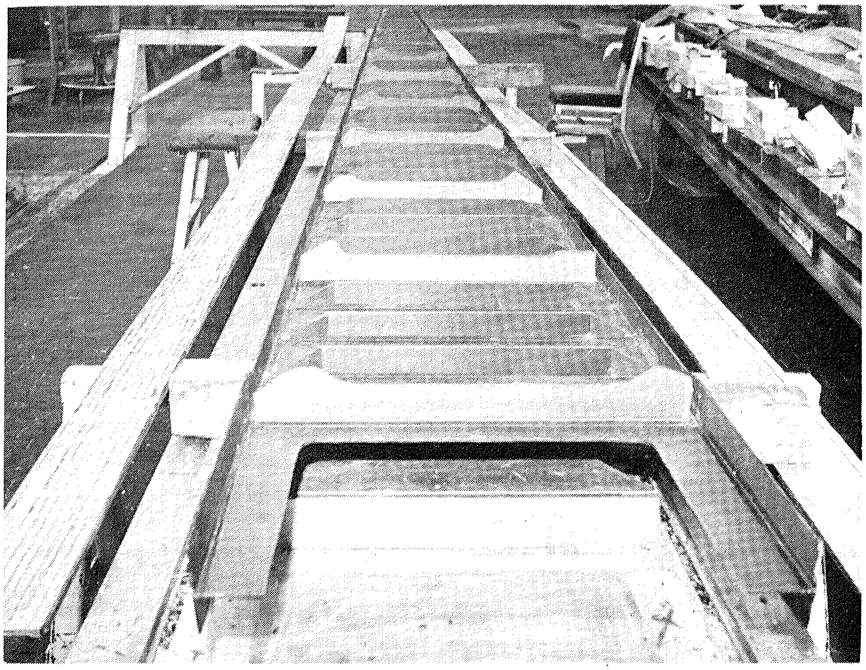
RL 3938-13

Figure 4-14. Spar Lifted Out and Turned Over



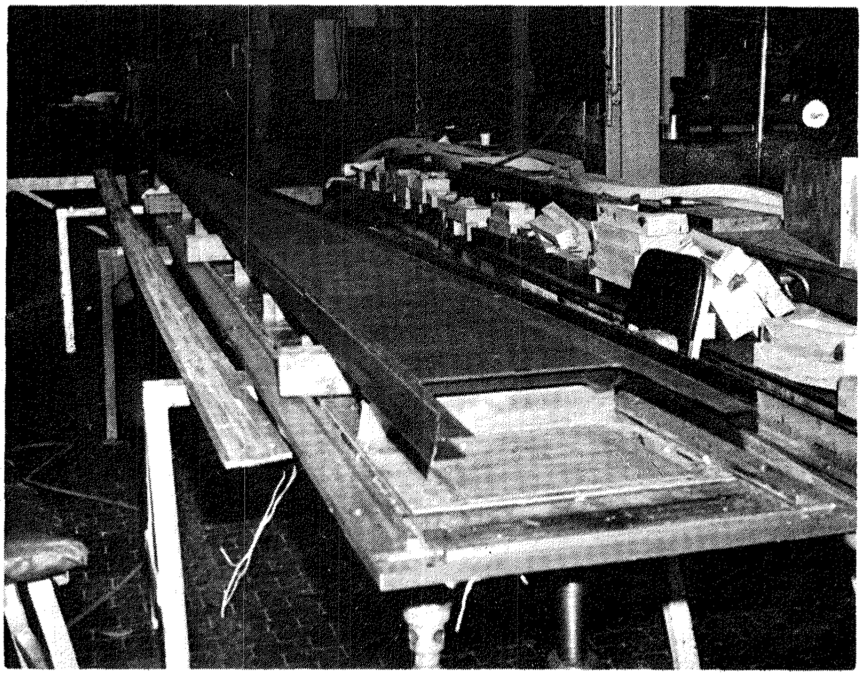
RL 3917-19

Figure 4-15. Spar as Removed from Tool (Upper End)



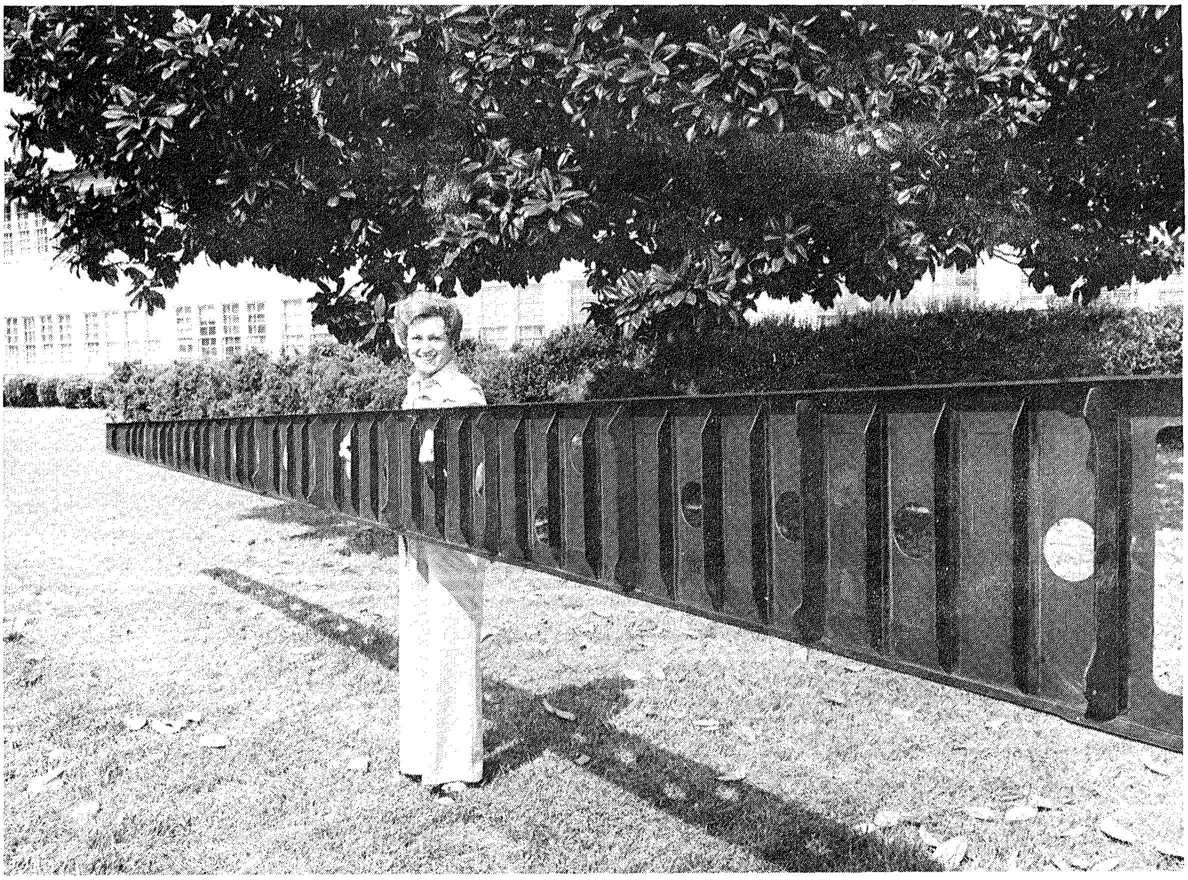
RL 3917-14

Figure 4-16. Spar as Removed From Tool (Lower End)



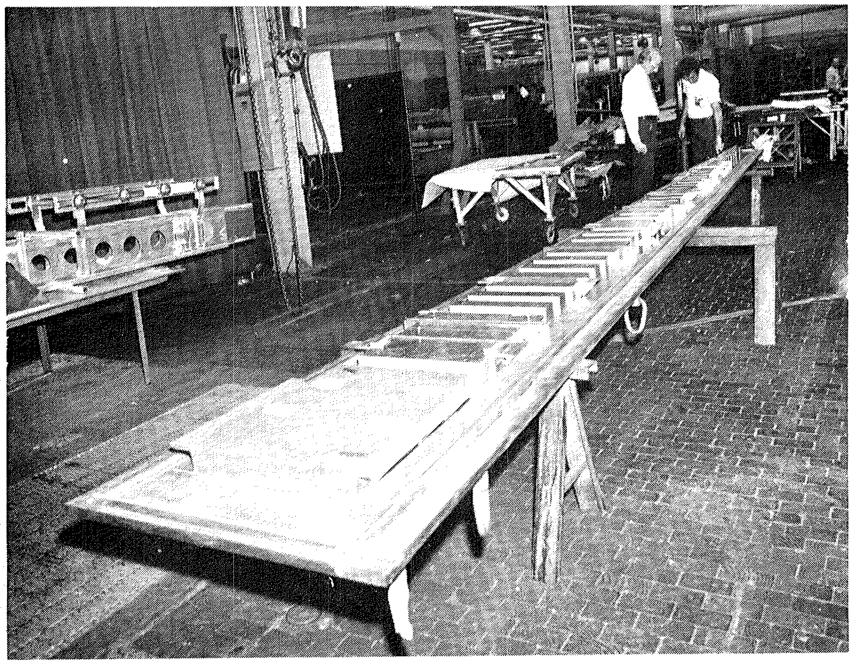
RL 3917 - 19

Figure 4-17. Spar As Removed From Tool (Fwd Side)



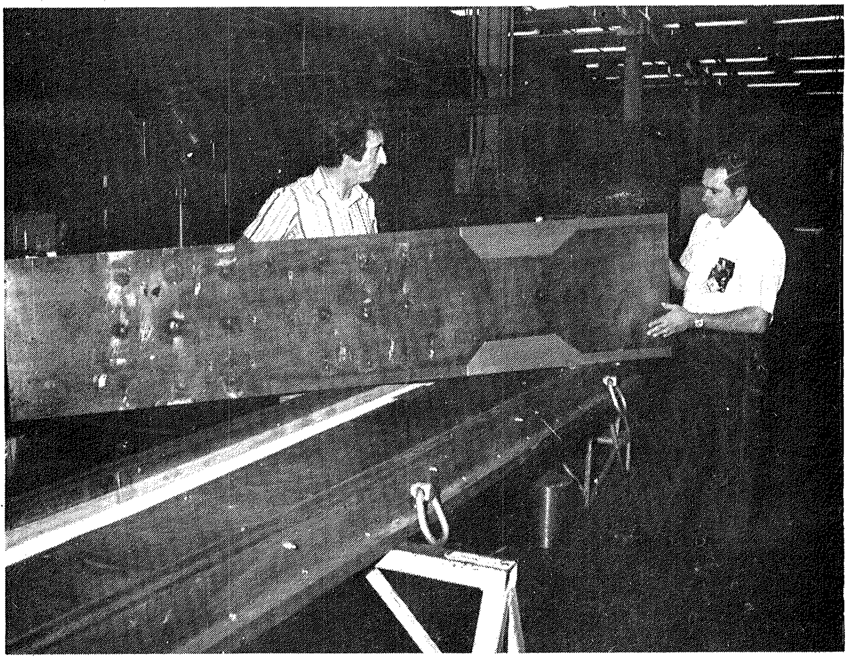
RL 3536-8

Figure 4-18. Aft Face of Completed Front Spar



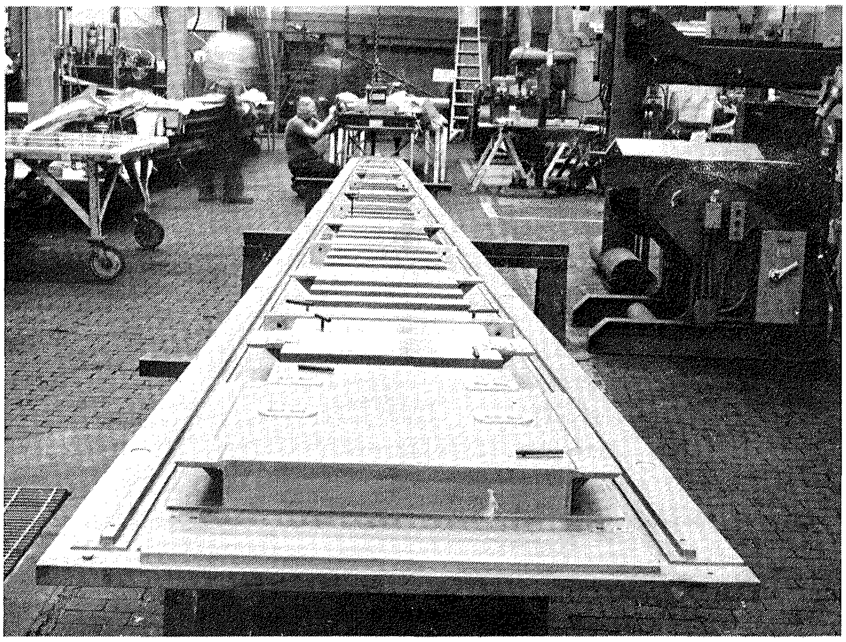
RL 4330-2

Figure 4-19. Island Blocks Placed on Base Plate



RL4337-2

Figure 4-20. Dummy Web Positioned In Tool



RL 4344-5

Figure 4-21. Island Blocks on Base Plate

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