NASA-CR-165,885

NASA Contractor Report 165885

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

PHASE IV— FINAL REPORT MANUFACTURING DEVELOPMENT

F. Alva, G. Brozovic, B. Carll,
R. Eudaily, J. Henkel, A. Jackson,
R. Johnston, B. Moseian, R. O'Brien

LOCKHEED-CALIFORNIA COMPANY
BURBANK, CALIFORNIA

CONTRACT NAS1-14000
MAY 1982

National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665

FOREWORD

This report was prepared by Lockheed-California Company and Lockheed-Georgia Company under contract NAS1-14000, Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft. It is the final report for Phase IV - Manufacturing Development activity covering work completed between 1 July 1977 and 1 June 1981. This 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. Herman L. Bohon is Project Manager for NASA Langley. The technical representative for NASA Langley is Dr. Herbert A. Leybold.

Engineering Development activity (Phase I) has been reported previously in NASA CR-144986, and Design activity (Phase II) has been reported previously in NASA CR-165634. Subsequent phases still to be reported are the Production Readiness Verification Tests (Phase III) and Ground Test (Phase V).

The following Lockheed personnel were principal contributors to the program during Phase IV.

Lockheed California Company:

Engineering:

A. M. James A. C. Jackson	-	Engineering Managers
D. C. Novelli	-	Materials and Processes
D. R. Pascal	-	Design
R. R. Van Cleave	-	Stress
J. Van Hamersveld	-	Producibility
R. H. Johnston	-	Cost
W. E. Newcomb	-	Cost

Manufacturing:

J. V. Henkel R. A. Short	-	Manufacturing Managers
B. L. Carll	-	Tool Design
F. Alva	-	Tool Design
G. R. Brozovic	-	Manufacturing Research
J. V. Henkel	-	Planning

Quality Assurance:

B. Mosesian	-	Quality Assurance Manager
J. F. Crocker	-	Nondestructive Test
F. W. Diggles	-	Quality Assurance

Lockheed-Georgia Company:

W. E. Harvill	-	Program Manager

Engineering:

R. R. Eudaily	-	Engineering Manager
R. O'Brien	-	Cost

Manufacturing:

R. B. Cantley	-	Manufacturing Manager
T. N. Bridges	-	Tool Design
W. W. Barber	-	Fabrication
R. H. Kilpatrick	-	Fabrication

Quality Assurance:

G. A. Swain	-	Quality Manager

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

TABLE OF CONTENTS (Continued)

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Continued)

LIST OF ILLUSTRATIONS (Continued)

LIST OF TABLES

ADVANCED MANUFACTURING DEVELOPMENT OF A COMPOSITE EMPENNAGE COMPONENT FOR L-1011 AIRCRAFT PHASE IV FINAL REPORT

BY

F. Alva
G. Brozovic
B. Carll
R. Eudaily
J. Henkel
A. Jackson
R. Johnston
B. Mosesian
R. O'Brien

SUMMARY

This is the final report on Manufacturing Development (Phase IV) conducted on the Advanced Composite Vertical Fin (ACVF) program. The significant elements of this program phase include the manufacturing and tool engineering plans, tool manufacturing, production tool proving, component manufacture, cost analysis, and quality control functions.

The manufacturing and tool engineering concept concentrated on developing an economical plan to manufacture three shipsets of the ACVF. The tool manufacturing used the tool engineering and manufacturing plan to produce production quality tooling and to develop data necessary to produce tooling capable of manufacturing graphite composite primary aircraft structural parts. Production tool proving was used to refine the manufacturing process specifications, as well as the tooling components, to provide flight-quality hardware.

Component manufacturing provided the necessary technical data to develop manufacturing cost analyses to update production cost projections. Enough components were produced to assemble two complete ACVF's. These components were manufactured in a production environment using production tooling in accordance with process bulletins and specifications.

The recurring costs analyses were updated and evaluated during this program phase. The Quality Control organization maintained records demonstrating traceability of materials and parts and provided ongoing manufacturing support by conducting and documenting nondestructive inspections and quality assurance and control tests.

INTRODUCTION

This ACVF program is part of the Aircraft Energy Efficiency (ACEE) Composite Structures Program. The broad objective of the ACEE program is to accelerate the use of composite structures in new aircraft by developing technologies and processes for early progressive introduction of composite structures into production commercial transport aircraft. This program, one of several which are collectively aimed toward accomplishing that goal, has the 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 operate safely and economically under service loads and environments, and that it will meet Federal Aviation Administration (FAA) requirements for installation on commercial aircraft. A limited quantity of units have been fabricated to establish manufacturing methods and costs. The advanced composite vertical fin (ACVF) uses advanced composite materials to the maximum extent practical and weighs 28 percent less than the metal fin it replaces. A method was 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 developed under this program consists 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 7.62 m (25 ft) tall with a root box chord of 2.74 m (9 ft) and represents an area of 13.94 m^2 (150 ft^2).

The primary objective 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 properly assess their cost effectiveness.

Lockheed-California Company, as the prime contractor, has overall program responsibility and has teamed with the Lockheed-Georgia Company in the development of the ACVF. Lockheed-California designed and fabricated the covers and the ribs, and is conducting the Production Readiness Verification Test (PRVT) program and the full-scale ground tests. Lockheed-Georgia designed and fabricated the front, rear, and stub spars, and has assembled the composite fin at their plant in Meridian, Mississippi, where the present L-1011 vertical fins are assembled.

The duration of this program is 77 months, with completion scheduled for June 1983. The master schedule is shown in figure 1. The program is organized in four overlapping phases: Phase II, Design and Analysis; Phase III, Production Readiness Verification Tests (PRVT); Phase IV, Manufacturing and Development; and Phase V, Ground Tests. Phase I, Engineer-Development, was completed in 1976; Phase II, has been completed; Phase IV Manufacturing Development has been completed and is reported herein; and Phases III, and V are currently in progress.

Production Readiness Verification Tests (Phase III) are designed to answer the following questions:

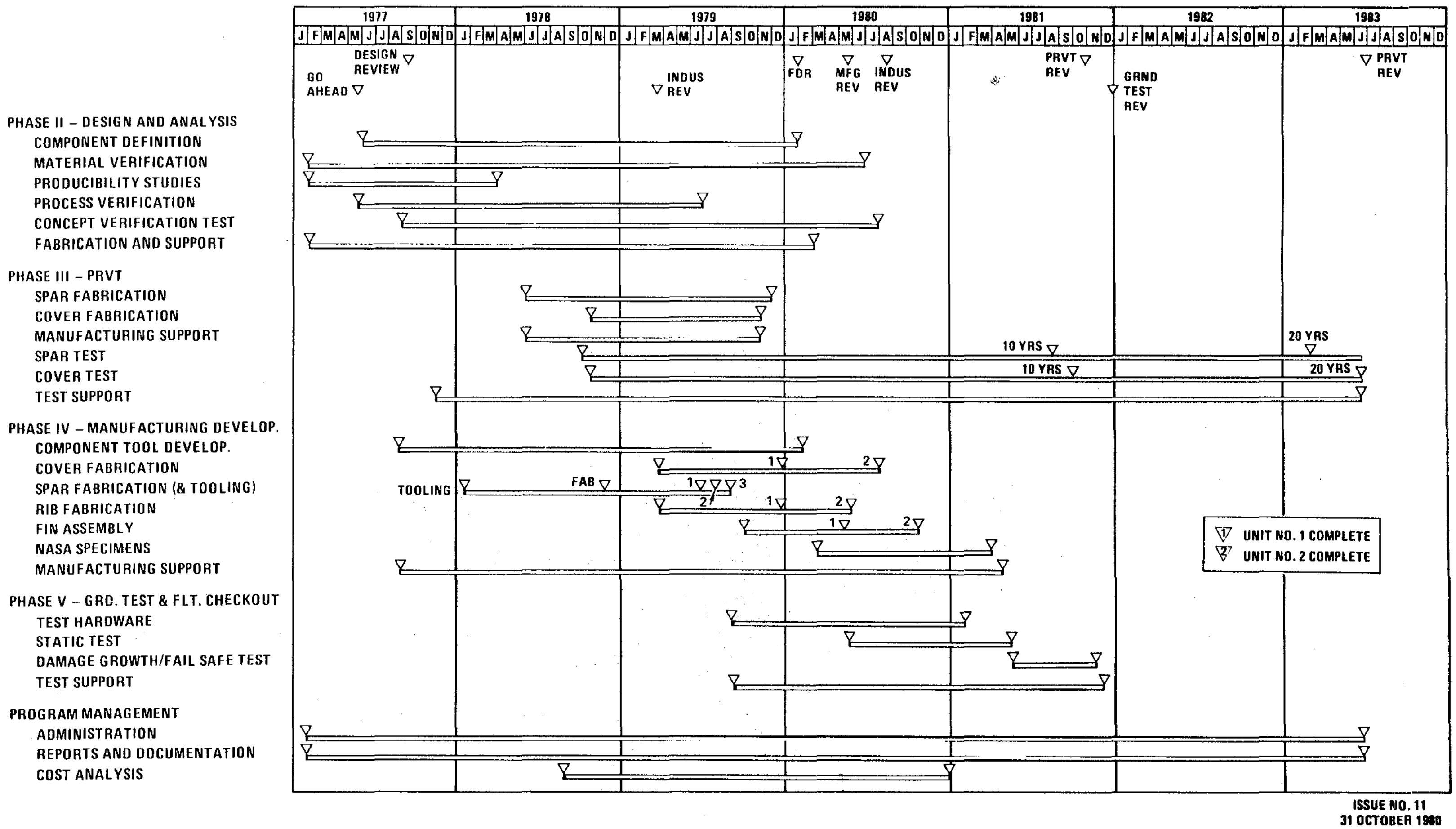


Figure 1. - ACVF program, master schedule.

- What is the range of production quantities 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? Are the margins sufficient to account for this variability?

- Will production quality components survive extended-time laboratory fatigue tests involving both load and environmental simulation of sufficient duration and severity to provide in-service confidence?

Ten static strength tests have been conducted and twelve durability tests are being conducted on each of two key structural elements on the ACVF. One element represents the front spar/fuselage attachment area, and the other element represents the cover/fuselage joint area. Two of the covers and two of the spars are being durability tested at strain levels 1.5 times those in the basic program comparable to strains used in primary structures such as horizontal stabilizers and commercial airplane wings. The satisfactory completion of these tests would permit a production commitment to be made without performing long term flight service testing and evaluation.

Ground tests will be conducted on one full-scale fin box beam structure mounted on simulated fuselage support structures during Phase V. The test plan will include static ultimate load, damage-growth test to one lifetime and fail-safe tests. Inspection and repair techniques for in-service maintenance will be employed throughout the tests. Test results will be used to verify the analytical, design, and fabrication procedures, and are essential imputs to the FAA for certification. Certification will be based on satisfying both static strength and damage tolerance requirements.

Throughout this program, technical information gathered during performance of the contract is being disseminated throughout the aircraft industry and to the government through quarterly reports that coincide with calendar quarters and final reports at the completion of each phase. All test and fabrication data are being recorded on Air Force Data Sheets for incorporation in the Air Force Design Guide and Fabrication Guide for Advanced Composites. Oral reviews have been conducted to acquaint industry and government with progress of the program.

Use of commercial products or names of manufacturers in this report does not constitute official endorsement of such products or manufacturers, either expressed or implied, by the National Aeronautics and Space Administration. Measurement values are stated in SI units followed by customary units in parenthesis. All work was performed using customary units.

1. COMPONENT DESCRIPTION

The fin box consists of 2 covers, 2 main spars, 1 stub spar and 11 ribs. Figure 2 shows an exploded view of the box.

1.1 Covers

The covers are designed primarily by stiffness. The fin box is designed to match the bending and torsional stiffness of the metal fin; as the root end has to match the existing joint to the afterbody; and all interfaces are unchanged. The cover skin tapers in steps from 34 plies at the root end to 16, 14, then 10. The edges are built up to 3.05 mm (0.12 in.), 24 plies, to allow for countersinking holes without feather edges. A thickness map is shown in figure 3 and the skin ply buildup is shown in figure 4. The 0-degree ply is oriented parallel to the rear spar.

The closed hat section stiffener was selected because of its torsional stability and the fact that it did not have to be tied to each rib. The stiffener spacing and the hat configuration are shown in figures 5 and 6. The spacing at the forward end is established by the stiffener runout. The stiffeners are terminated at the ribs adjacent to the front spar, and the hat flanges continue under the rib caps to minimize any tendency to peel. The spacing of the remaining stiffeners is dictated by the stub spar location.

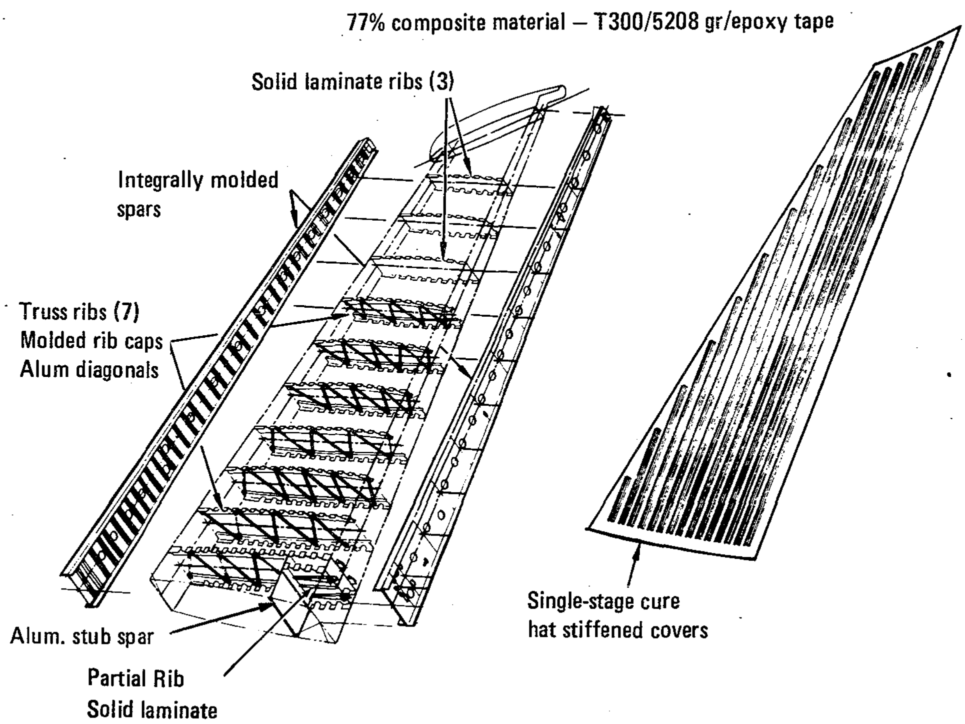


Figure 2. - ACVF design configuration.

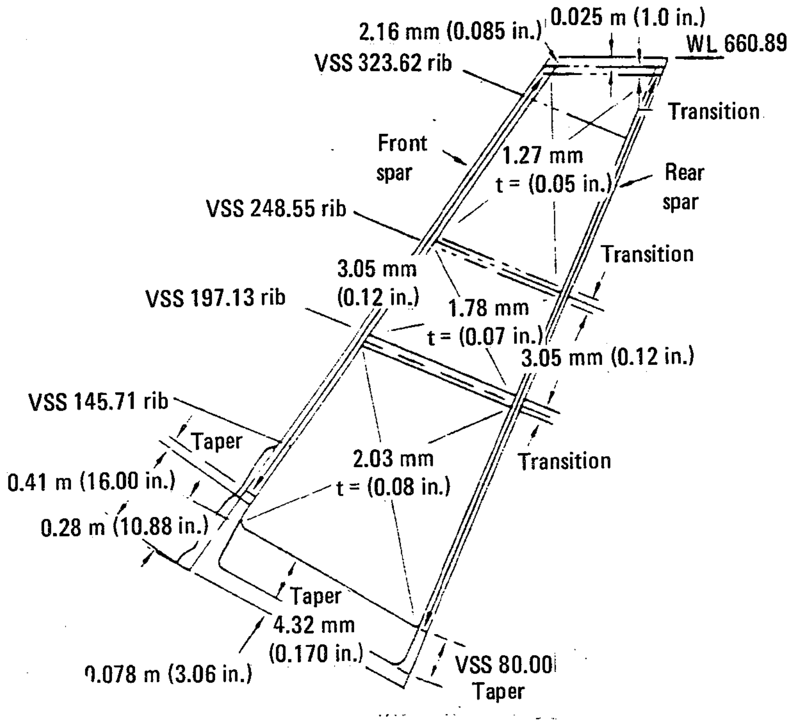


Figure 3. - Cover thickness.

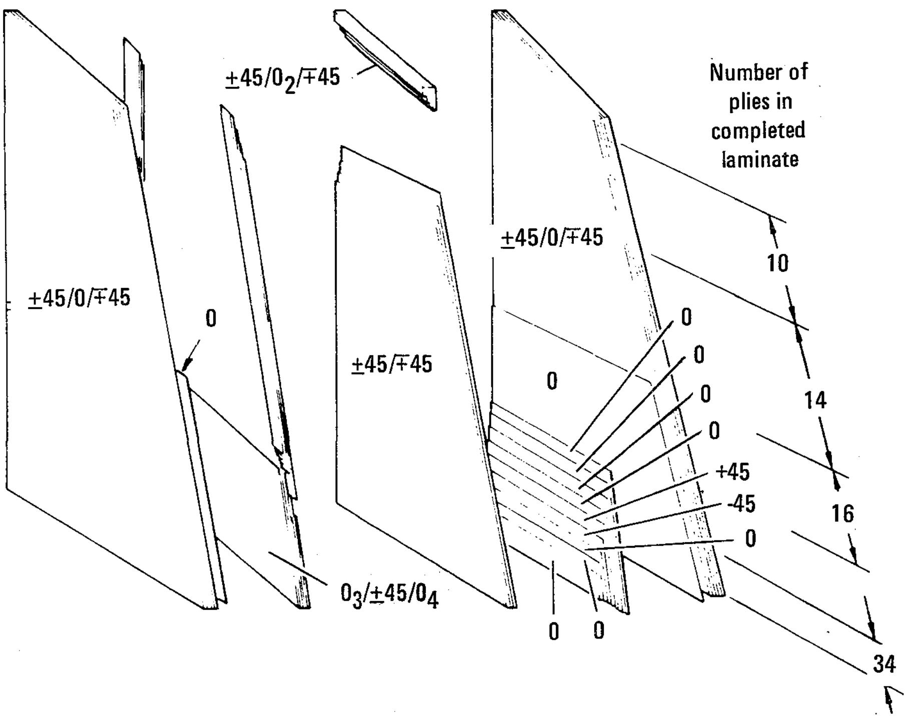


Figure 4. - ACVF skin ply buildup.

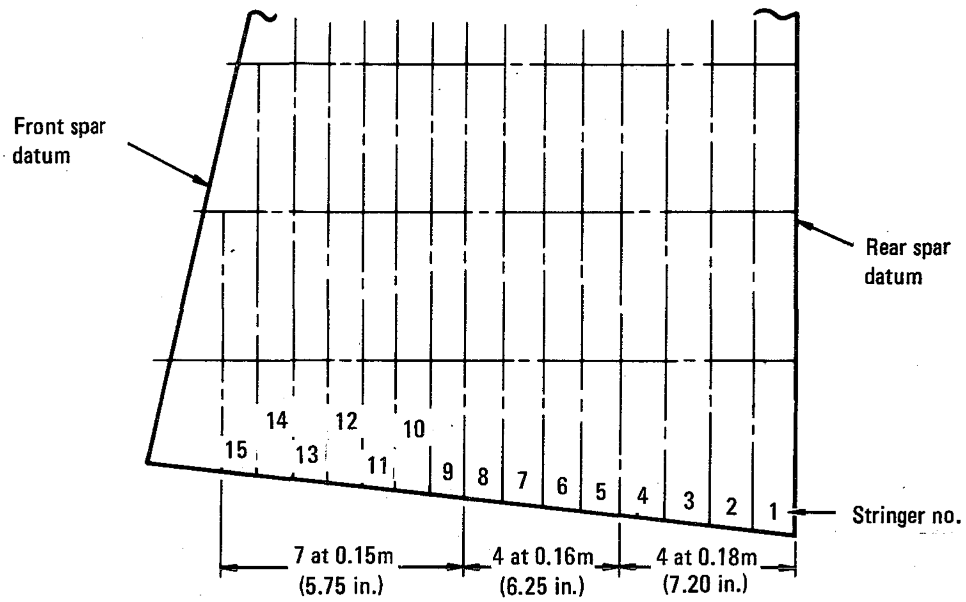


Figure 5. - ACVF stiffener spacing.

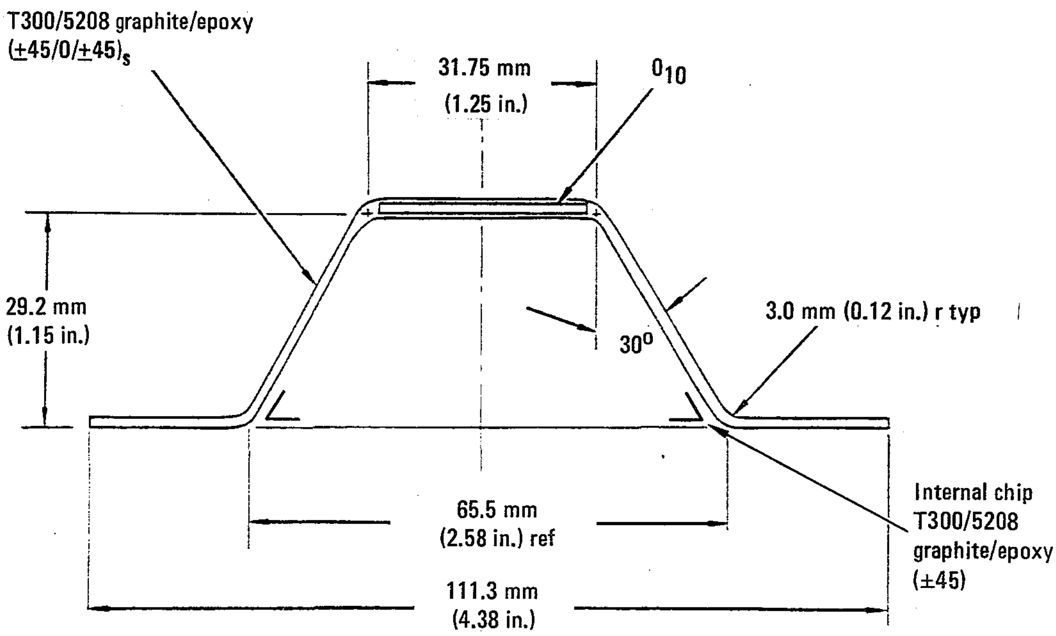


Figure 6. - Stiffener configuration.

The stiffener is built up of two 5-ply segments with a 10-ply segment sandwiched between them in the crown. A short segment of eight doubler plies is added only at the root end to stiffen the side walls for shearing out the crown loads. Internal clips consisting of two plies at ±45 degrees are added to prevent peel.

1.2 Ribs

The eleven ribs fall into three basic categories: The two lower ribs are actuator ribs, the next six are truss ribs, and the upper three are solid web ribs.

The actuator ribs consist of a partial solid graphite web at VSS 90.19 and a combination solid graphite web and graphite cap aluminum truss rib at VSS 97.19 shown in figure 7. The solid web is a 16-ply layup $(+45/0/-45/90_2/-45/0/+45)s$. The sides adjacent to the covers are flanged to provide part of the skin attachment. Additional cap is provided by a C-section consisting of a 19-ply layup $(\pm 45/90/\mp 45/0/\pm 45/\bar{0}_3)s$. This cap extends the full length on VSS 97.19. The forward portion of this rib consists of the graphite epoxy C-section caps and aluminum cruciform extruded truss members. The truss rib caps are C-section caps consisting of 19 plies with the same layup as the VSS 97.19 cap. The truss members are again aluminum cruciform extrusions. A typical truss rib is shown in figure 8.

The solid web ribs are a sandwich design. The fin box becomes too shallow near the tip to use the truss design efficiently. The most cost-effective design is one without stiffeners. Because of the size of the rib web, an all graphite-epoxy shear buckling resistant design would be heavy. Thus, a syntactic epoxy core is used. Syntactic epoxy is an epoxy system filled with glass microballoons which has about half the density of graphite-epoxy. The face sheets consist of seven plies laid up as $\pm 45/0/90/0/\mp 45$. The edges around the core are graphite epoxy laid up as $\pm 45/0_2/\pm 45$. The uncured syntactic core is 0.95 mm (0.0375 in.) thick and compresses down to about 0.76 mm (0.03 in.) during cure. The configuration of the solid web rib is shown in figure 9.

1.3 Spars

Front and rear spars have been designed to comply with overall program objectives of providing at least a 20-percent weight savings over the metallic design, while maintaining production costs and ensuring structural and functional interchangeability with the baseline article.

The design concepts selected are the graphite-epoxy configurations shown in figures 10 and 11. The front and rear spars are similar in shape and size and are basically one-piece components with rib attach angles, stiffeners, caps, and webs integrally molded in a single cocured operation. The front spar cap forward flange, rear spar cap aft flange, and the fuselage joint areas have been configured to interface with the existing metallic

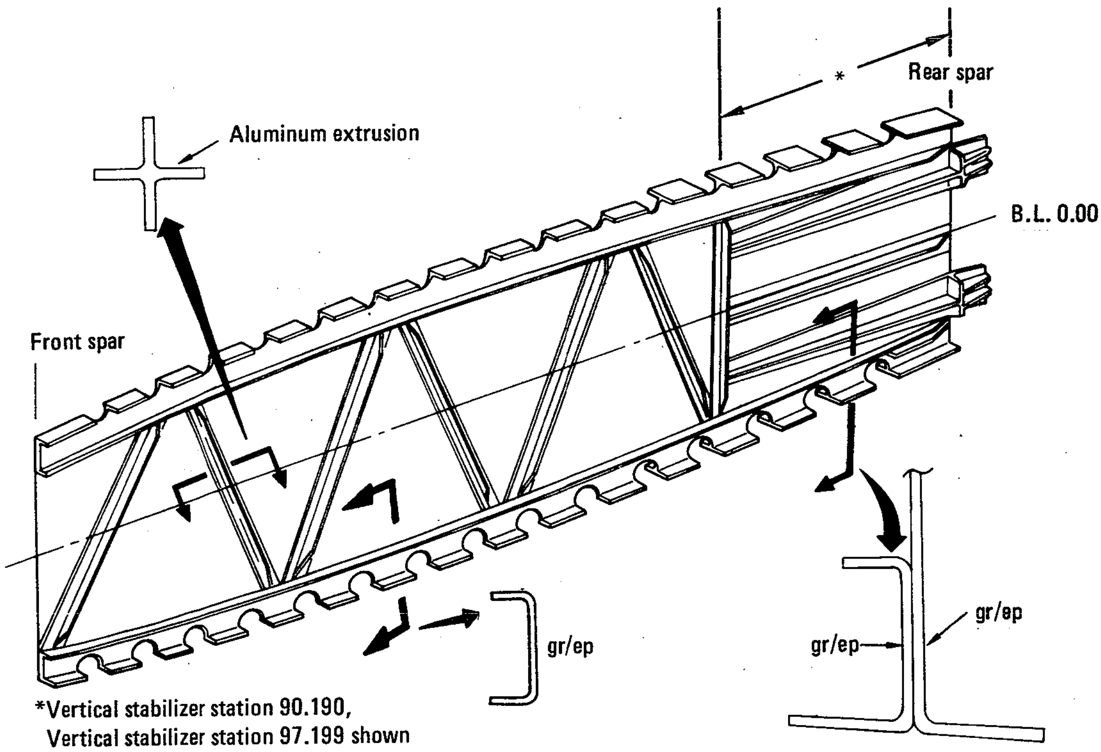


Figure 7. - Actuator rib.

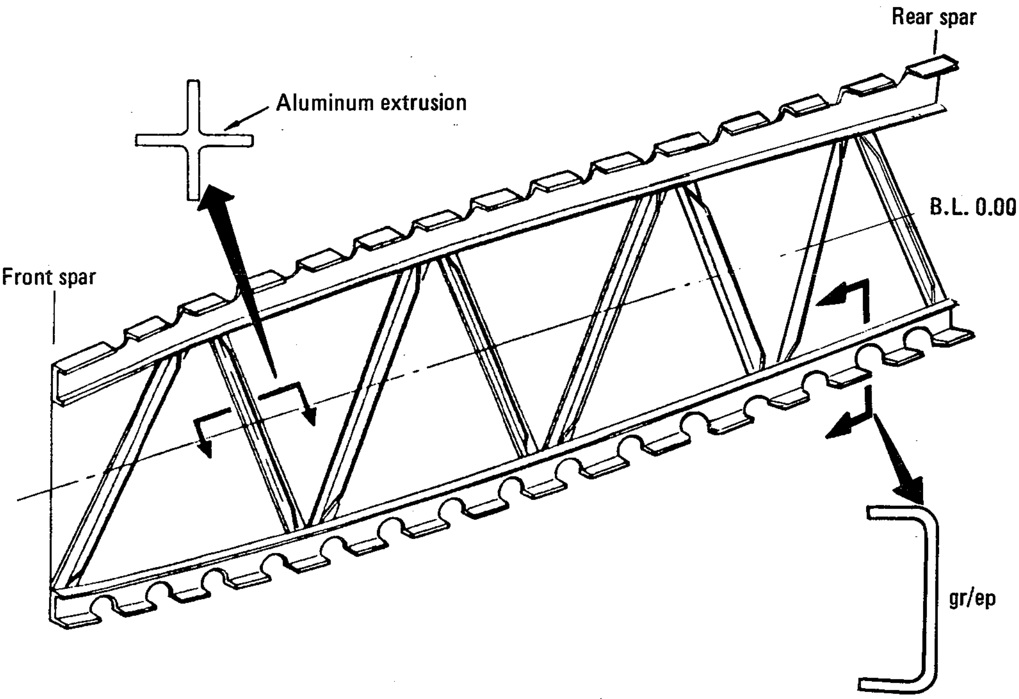


Figure 8. - Truss rib design.

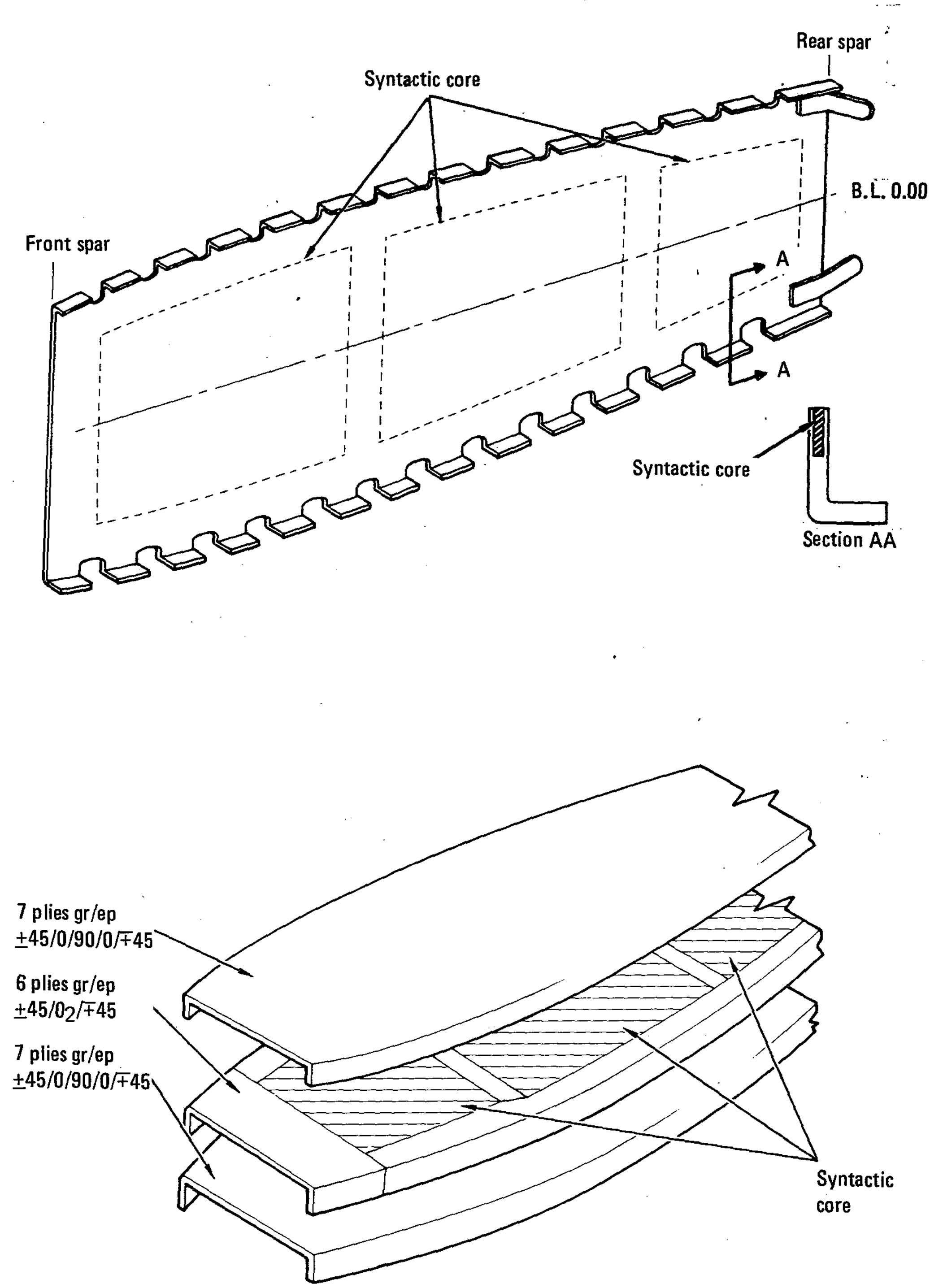


Figure 9. - Typical solid web rib design.

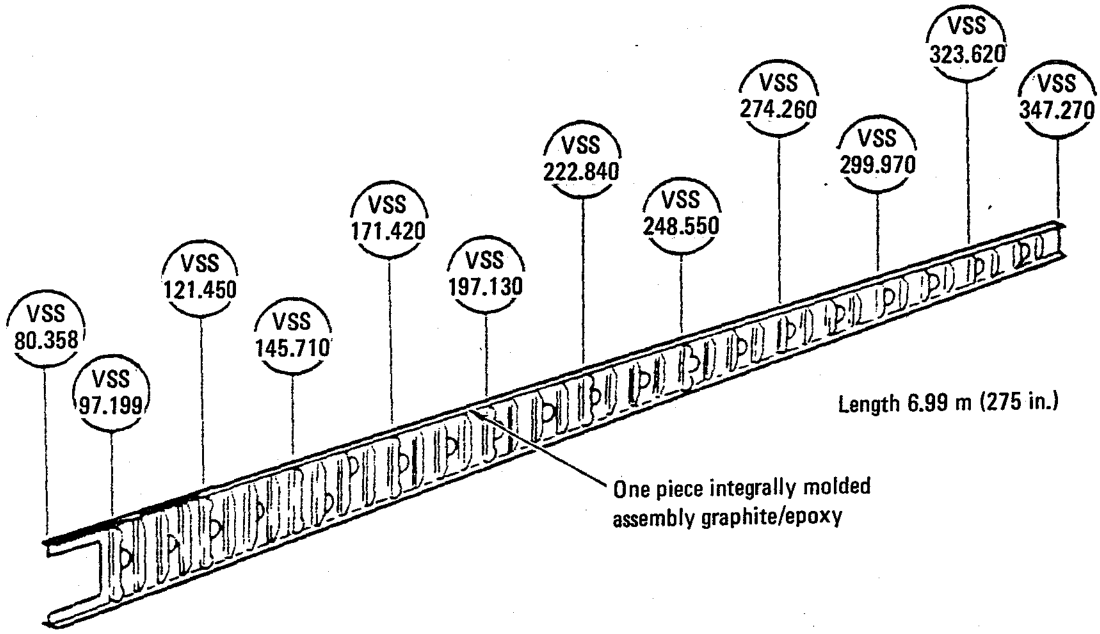


Figure 10. - Front spar assembly.

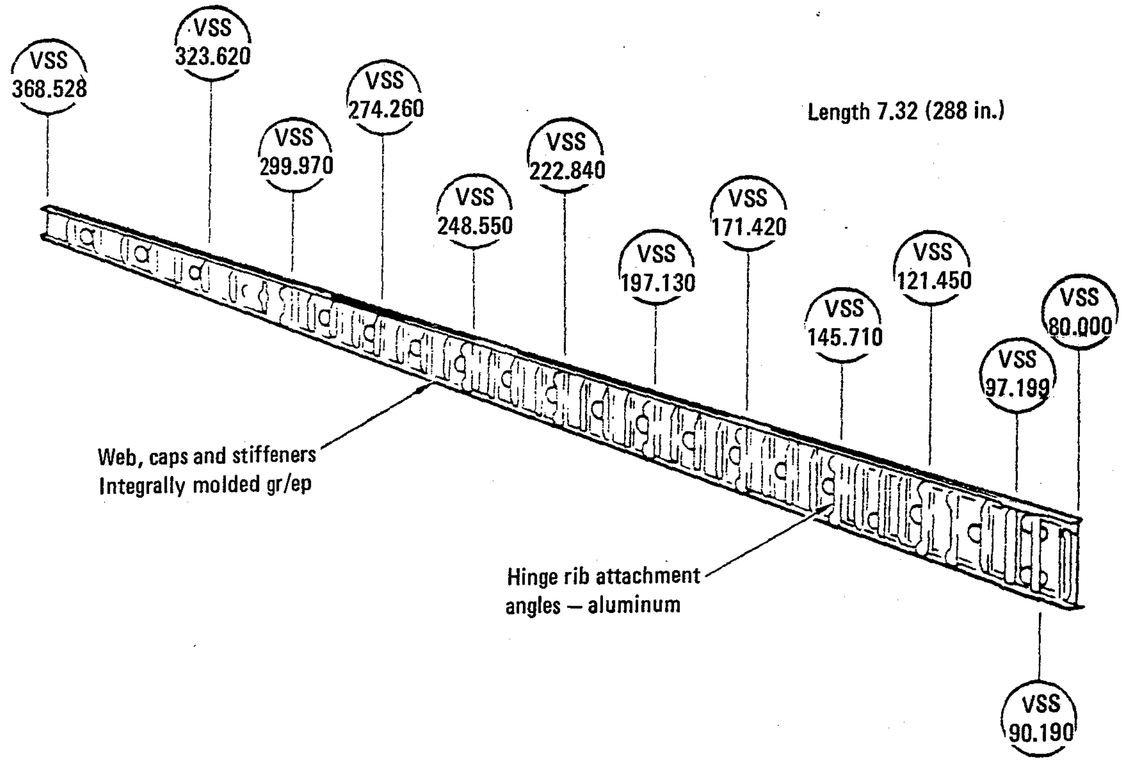


Figure 11. - Rear spar assembly.

structure and, therefore, do not necessarily represent the most efficient designs for advanced composite structures. Another critical interface area is the attachment of rudder hinges to the rear spar. To ensure that these locations are accurately maintained, separate aluminum attachment angles are jig located on assembly and mechanically attached to the spar.

Strength and stiffness requirements are controlled by selecting ply layups with a sufficient number of ±45-degree plies in the webs to provide the required shear strength and 0-degree plies in the caps for axial loading. To facilitate fastener installation in the final assembly fixture, access holes have been provided in the spar webs. Two access holes are required in each rib bay, and this dictates that three web stiffeners are added between ribs to ensure uniform hole spacing. The access hole edges are not reinforced.

1.3.1 Stub spar. - The stub spar shown in figure 12 is located between the aft fuselage closure rib and the rudder actuator rib, and has been retained as an aluminum assembly. This component cannot be attached until all fasteners in the fin-to-fuselage joint are installed. Therefore the component must be fabricated in small sections capable of passing through the root rib access hole. This involves a considerable amount of drilling and assembly of details on location, which were major factors in the decision to avoid using composite materials. Minor changes have been made to this component with respect to redesigned ribs and cover assemblies.

1.4 Box Assembly

Vertical fins for the ACVF program were assembled at the Lockheed-Georgia Company facility in Meridian, Mississippi, using an existing assembly fixture suitably modified to accept the various advanced composite components. Use of this fixture (where rudder hinges, rudder actuator, and fuselage attachment control points have been retained) will ensure that all interchangeability requirements are met.

The fin box assembly is illustrated in figure 13. Parts of the skin are cut away to show details of cover hats, ribs, spars, and joints used to assemble the L-1011 ACVF box. The fasteners selected for the assembly of major components are titanium hi-loks with stainless steel collars, which are wet-installed with sealant in close-tolerance, noninterference-fit holes.

Access to the inside of the box is accomplished by the removal of rib truss members and entry from the fuselage joint area. Limited hand access is also available through the holes provided in the front and rear spar webs. This access allows hi-loks to be installed at approximately 95 percent of all fastener locations and at the remainder, blind fasteners are used.

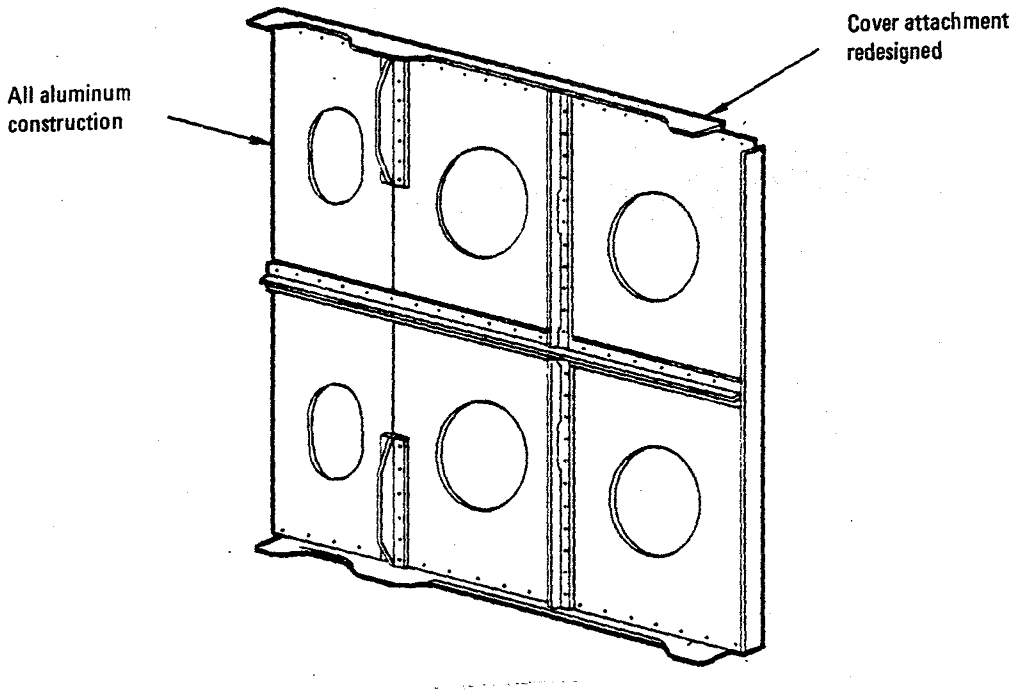


Figure 12. - Stub spar assembly.

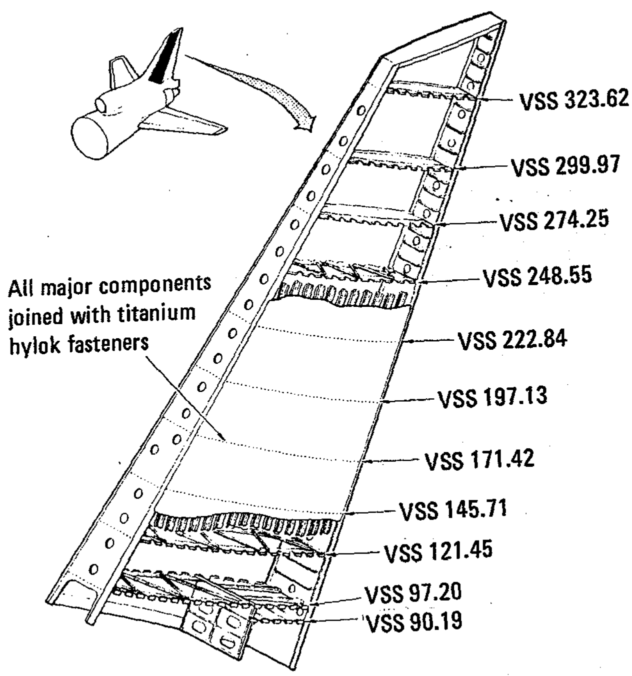


Figure 13. - Fin assembly.

Ribs, spars, and covers were designed to eliminate interference on assembly by assuming adverse tolerances at component interfaces. Where gaps in excess of 0.25 mm (0.010 in.) exist, Kevlar shims of the approximate thickness were installed.

2. COVER ASSEMBLY MANUFACTURING DEVELOPMENT

2.1 Tool Engineering Concept

2.1.1 Tool design features

- Interchangeability: The cover assemblies are not classified as interchangeable, so no consideration was necessary for interchangeability.

- Critical interfaces: Although none of the fit relationships of the cover assembly to the aircraft afterbody and box assembly rib caps required exceptionally tight control, the following features were controlled to ensure interference-free assembly and installation capabilities.

  - The relationship between the root end skin step and the end of the hat stiffeners on the cover assembly was controlled to fit the root rib of the afterbody. This was accomplished using a positive located caul plate in the cover assembly bonding fixture.

  - Rib cap to cover assembly hat stiffener clearances were controlled by coordinating the rib cap trim tooling to the hat stiffener cauls in the cover assembly bonding fixture using master tooling templates.

- Tool manufacturing cost: The cost of cover cure tooling was held to a minimum by designing a female cover assembly bonding fixture, which made it possible to use a one-piece contoured tool surface that did not require machining as a male tool would require various level caul surfaces to support the hat stiffeners, fillers, doublers in addition to the skin layup. Additional advantages are contoured external skin surface smoothness and a less complex hat stiffener internal support system during cure.

2.1.2 Tool manufacturing process.- The all-steel tool framework consists of channels with Macomber-type beams extending upward to rectangular tubing. Angles are mounted on the tubing to hold contour plates in the proper vertical stabilizer station location. The contour plates were numerically control machined and oriented to minimize resin flow due to gravitational force.

The faceplate shown in view B and F of figure 14 is a one-piece 6.35 mm (0.25 in.) thick ATSM standard A36 hot-rolled plate, which is intermittent bead-welded to the frame to minimize warpage. The faceplate of the fixture required no preforming to conform to the loft contour of the contour plates.

Only slight pressure at the tip end was necessary using shot bags to cause the faceplate to lay to contour. Portions of the tubular framework of the tool were used as vacuum manifolds. Vacuum outlets were installed in the faceplate around the periphery of the cover assembly. Permanent-type thermocouples were installed on the underside of the faceplate.

Inflatable molded rubber mandrels (open to autoclave atmosphere) were developed and fabricated by Manufacturing Research to provide internal support and pressure application for the cover assembly hat stiffeners against the tooling hat caul plates. Cover assembly hat stiffener caul plates were constructed by hydraulically formed sheet steel. These cauls were located on the bonding fixture in proper relationship using machined steel spacer bars between the cauls at every other rib station as shown in figures 14 and 15.

2.1.3 Tool proving.- Some of the more significant problems encountered during the production of full-scale tooling for the cover assembly program are discussed below.

2.1.3.1 Expansion and contraction differential: Examination of the bonding fixture during removal of the vacuum bag and bleeder system of a cured cover assembly indicated a differential of expansion and contraction existed between the composite part and the steel tool during the heat-up and cool-down of the autoclave cure cycle. It became evident the cover assembly and tool expanded approximately 9.65 mm (0.38 in.) at 412K (270°F). After cure, the tool contracted to its original length leaving the composite part 9.65 mm (0.38 in.) longer than the tool peripheral cauls permitted. This made it necessary to design and construct tool details that could control the periphery of the part during the cool-down cycle and would permit movement under slight pressure preventing part damage. In the case of the cover assembly, it was necessary to locate and fasten the bonding fixture root end caul plate with tubular rivets, which would shear during cool-down. When calculating expansion factors for graphite composite components this lack of contraction must be accounted for. The degree to which contraction occurs is directly affected by the graphite fiber orientation which varies from one part configuration to another. The effect of ply orientation on the thermal expansion is shown in figure 16.

2.1.3.2 Bleed resin adherence: A condition which did not directly affect the cover assembly, but proved to be a driving factor in increasing tool turnaround time, was bleed resin adhering to the bond fixture details during the cure cycle. The hat caul locating spacer details were exceptionally difficult to remove during the debagging process and, when removed, caused deformation and damage to the spacer locating pads on the hat caul plates. This made it necessary in some instances to set up all the master tooling templates on the bonding fixture and repair, relocate, or replace many of the spacer locating pads.

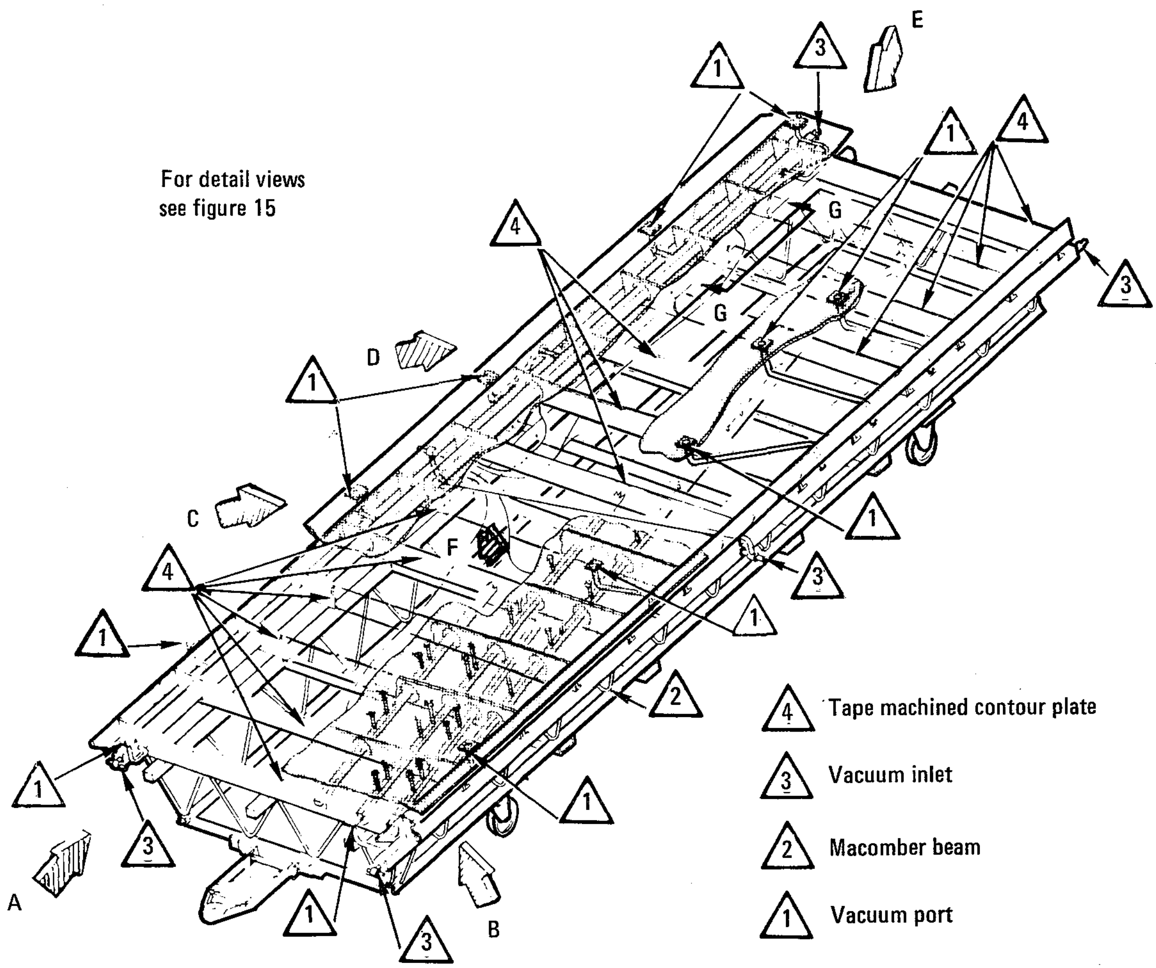


Figure 14. - Support frame of cover assembly bonding fixture.

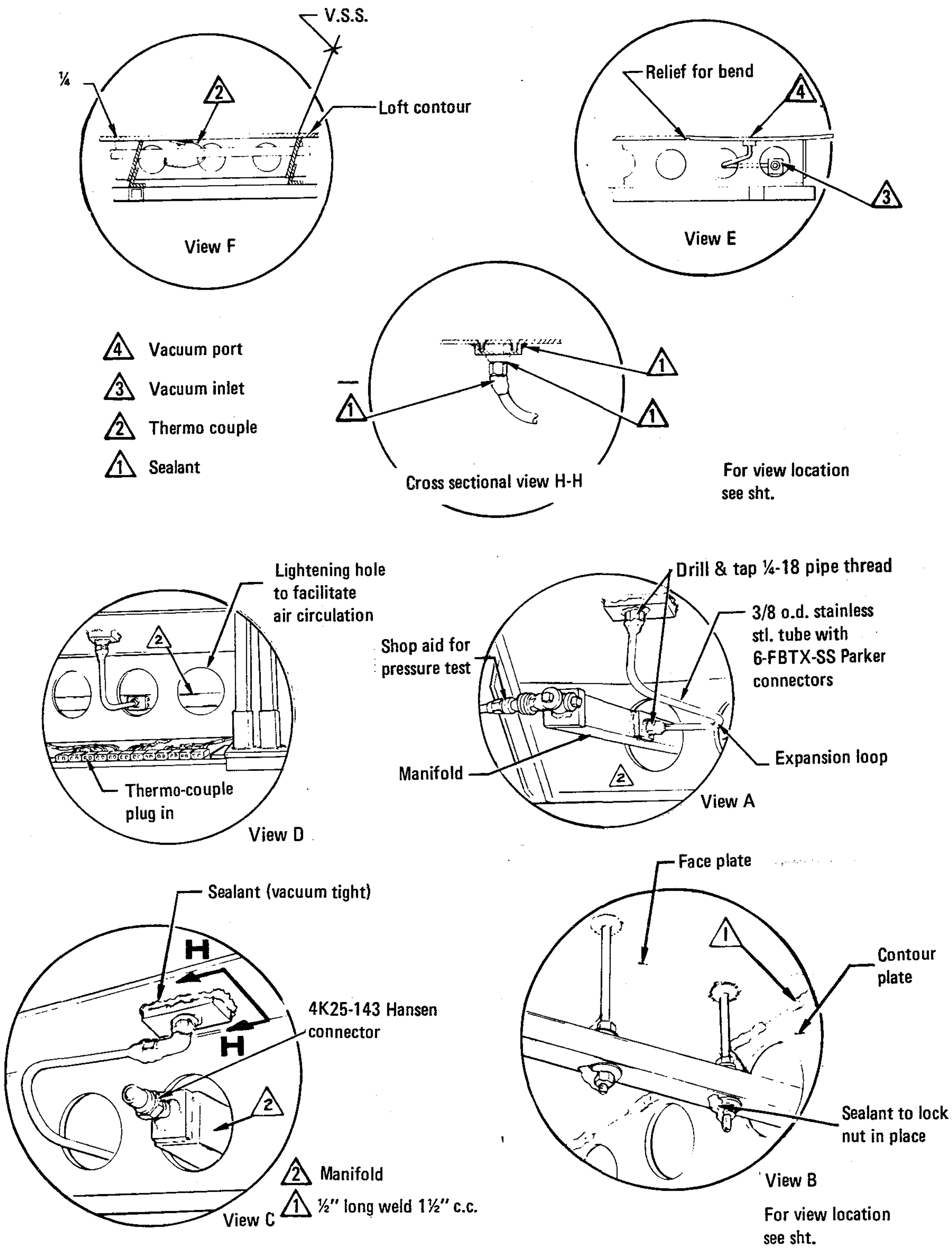


Figure 15. - Detail view of bonding fixture.

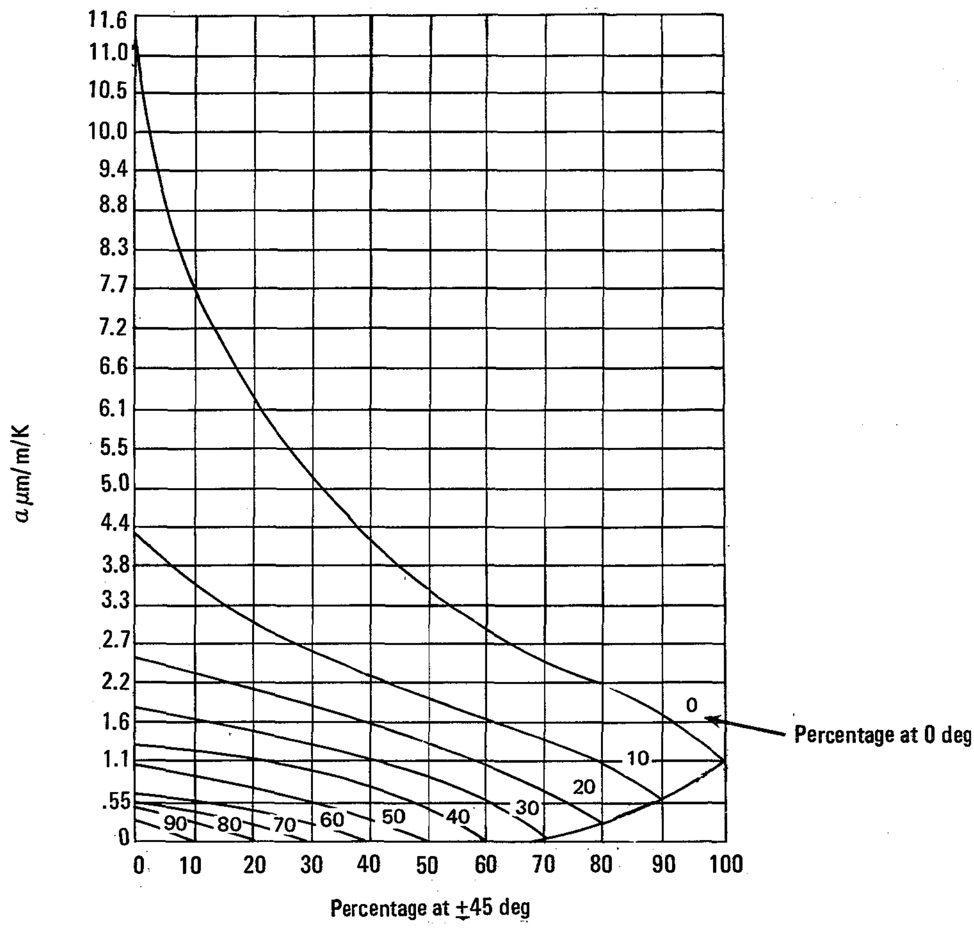


Figure 16. - Effect of graphite composite fiber orientation on expansion.

In addition to the standard application of an aerosol air dry multicycle halogen release system, which was originally applied to the tool and tool details, a system was needed to reduce the adherence of bleed resin. After testing several systems it was found that applying a baked-on Teflon-like coating to the tool details with a final coat of the halogen release system significantly reduced man-hours and turnaround time necessary to prepare the bonding fixture for a subsequent layup.

2.1.3.3 Graphite compaction: Dimensional inspection of the completed cover assemblies revealed the hat stiffeners had a tendency to drift out of proper location during the cure cycle. This was due to the cover assembly being laid up in a female tool with the hat cauls on top of the skin plies. As the skin plies were compacted during vacuum debulk and cure, the radius of the skin became larger. This caused the hat-locating details to have gaps between them allowing the hat cauls and hat stiffeners to shift during cure, figure 17. To reduce the effect of this condition, shims were placed between the spacer bars and the locating pads on the hat stiffener cauls subsequent to each vacuum debulk operation while the hat stiffeners were located on the skin layup, figure 18.

Although the theory was never proven due to restrictive autoclave availability schedules and out-time of material, it was the opinion of Manufacturing and Manufacturing Research that additional accuracy of hat locations after cure could be achieved if the layup was pressure debulked at ambient temperature in the autoclave and subsequently reshimmed.

2.2 Manufacturing Process Description

2.2.1 Covers.- The cover assemblies consist of the skin plies, doublers, fillers, and hat stiffeners that are laid up manually and cocured as follows:

- Cover fillers: The cover fillers are laid up in sheet size and trimmed to net size per a layout template. Each filler shop order produces one cover assembly requirement of fillers, which are identified, packaged, and stored in the freezer. The fillers consist of 10 and 18 plies, oriented at 45 and 135 degrees, alternately. The root end fillers are step tapered and contain eight additional plies.

- Cover hat stiffeners and reinforcing straps: The cover hat stiffeners are laid up on layup blocks using four preplied groups, figure 19. The hat stiffeners are then trimmed to net width using trim templates (ATTs). They are then identified, packaged, and stored in the freezer. The reinforcing straps are also preplied, trimmed by hand, and B-staged (B-staging is defined in process improvements). The preformed straps are identified, packaged, and stored in the freezer. The straps consist of a 45-degree and a 135-degree ply and are shown in figure 20.

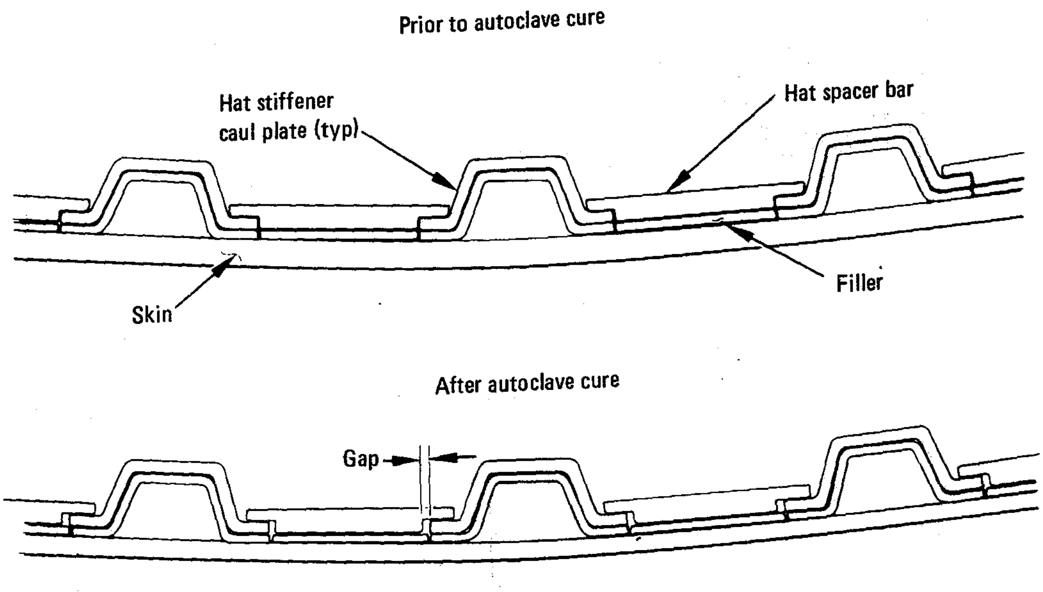


Figure 17. - Hat movement during cure.

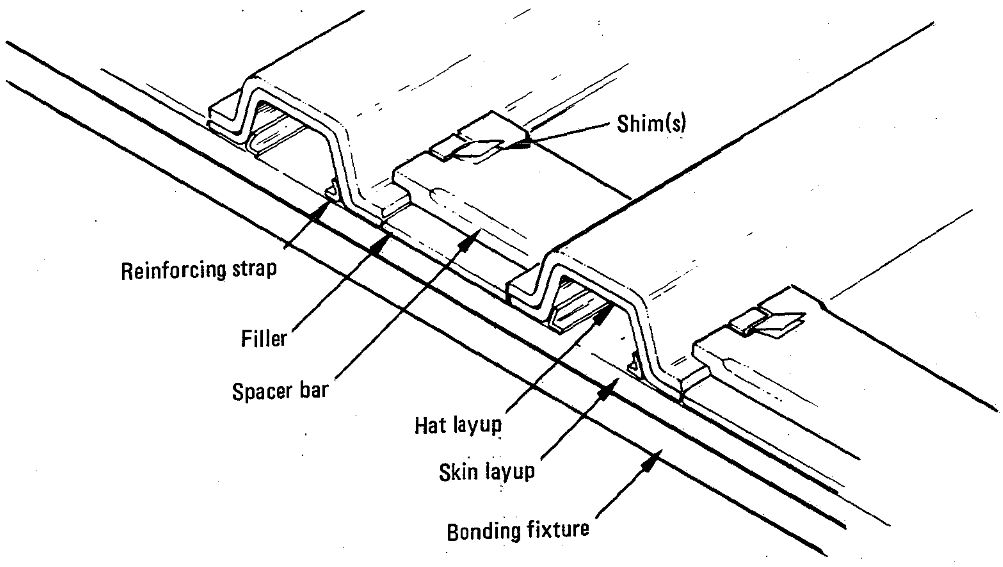


Figure 18. - Shimming to hold hat caul location during cure.

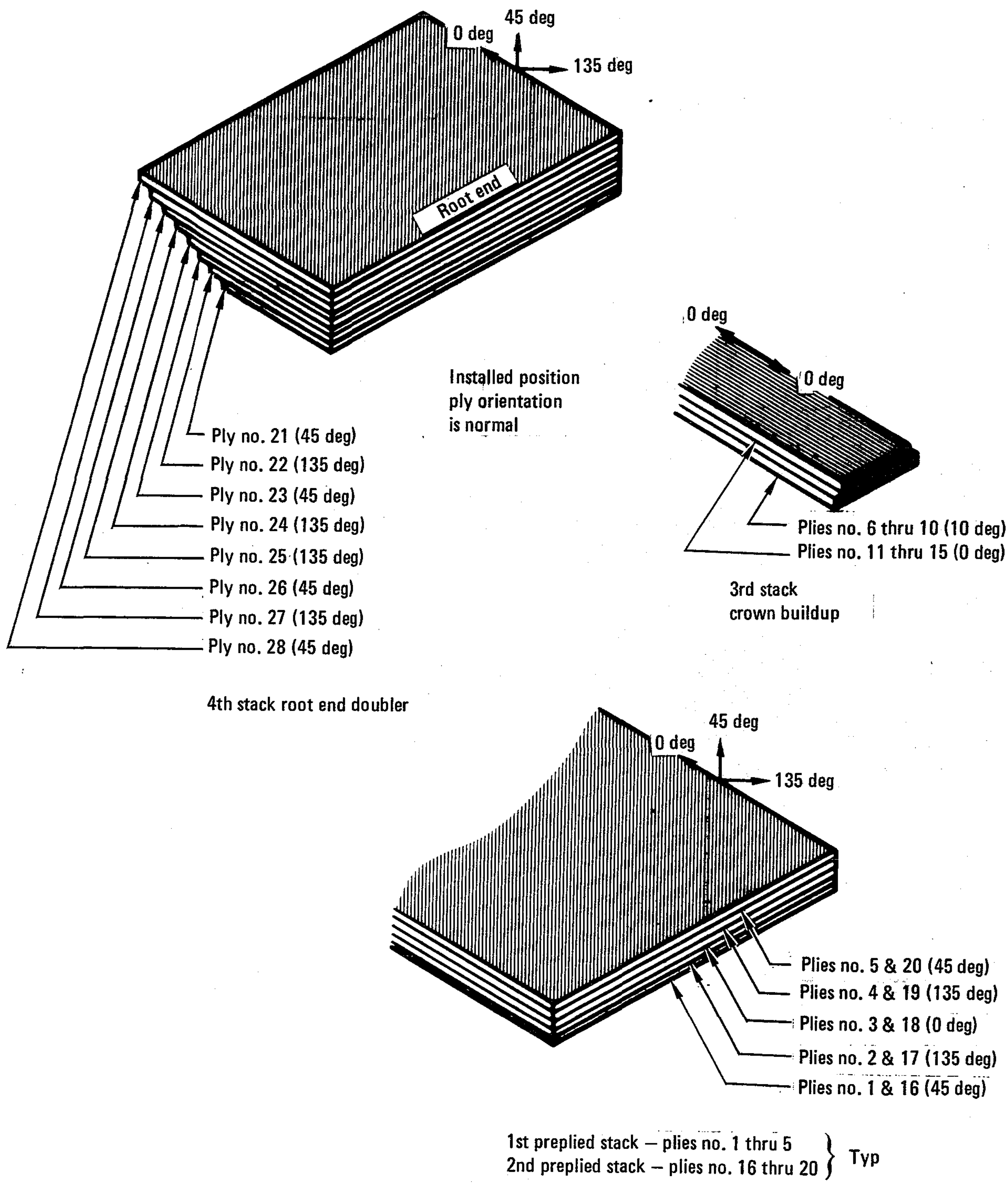


Figure 19. Hat stiffener preplied stacks.

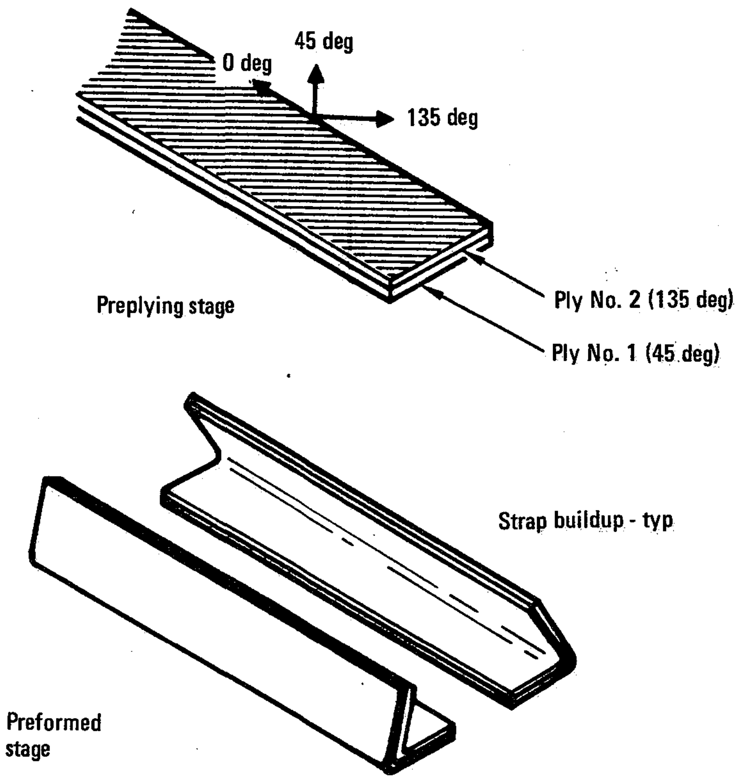


Figure 20. - Hat stiffener reinforcing straps.

- Cover assemblies: The cover doublers are preplied and laid up on the metal bonding fixture (MBF) with the skin plies. The hat stiffeners and straps are fitted on the mandrels, placed in the hat cauls, and the loaded cauls located on the MBF. The assembly is then bagged and subsequently cured in the autoclave.

2.2.2 Cure cycle.- The cure cycle was modified on the third assembly to comply with a mandatory safety requirement to maintain the oxygen content of the autoclave below a specific level for safe operating conditions. To meet this requirement, heat and pressure were applied immediately. When the autoclave reached 345 kPa (50 psi) and 339K (150°F), the oxygen content dropped to a safe operating condition. Part temperature, in the meantime, had risen to only 316K (110°F) maximum. A pressure leak check was made at time time. Pressure was reduced to 38 kPa (20 psi) and maintained while part temperature was raised to 372K (210°F). After dwell at 372K (210°F) was complete, part temperature was raised to 400K (260°F). After dwell at 400K (260°F) was complete, pressure was raised an additional 449 kPa (65 psi) to comply with minimum pressure of 586 kPa (85 psi). The remainder of the cure cycle followed the nominal requirements of the process bulletin, figure 21.

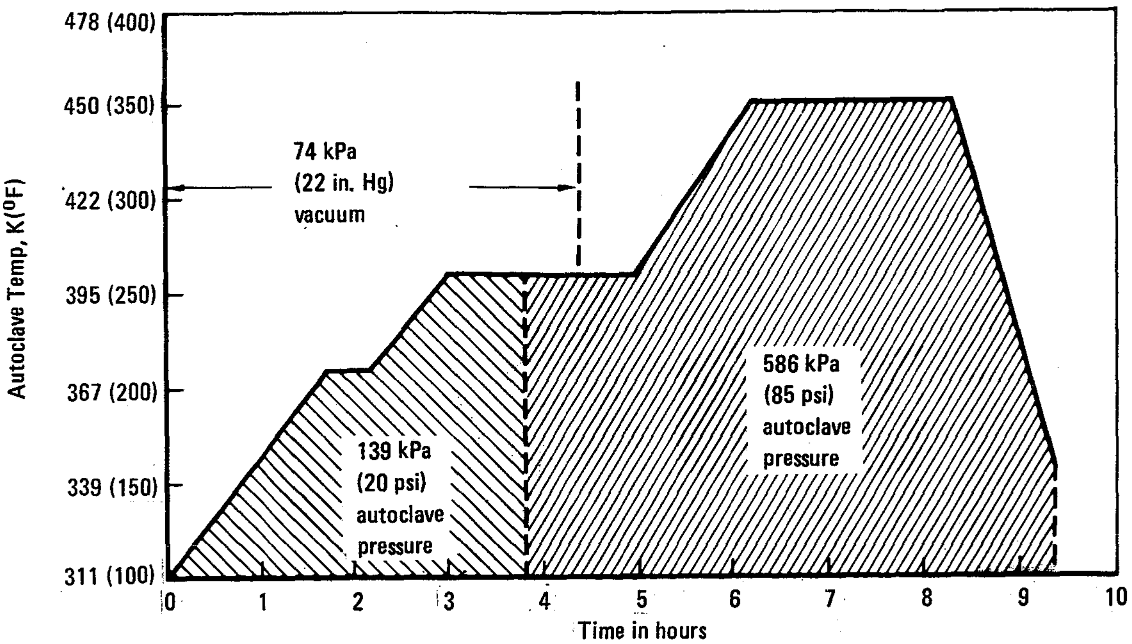


Figure 21. - ACVF covers, cure cycle.

2.2.3 ACVF cover assembly bleeder/breather stacking.- The bleeder stacking used on the cover was modified to reduce man-hours involved in this phase of the layup operation. All bleeder plies of nexus and 120 cloth, normally used over the skin area between hat stiffeners, were removed. The 1-ply strips of peel material normally used over the hat stiffeners were replaced by 2 plies of peel material. Since these strips also overlap each other in the skin area between hat stiffeners, the peel ply material between hats was doubled from 2 plies to 4 plies. These additional layers of peel material provided bleed characteristics equivalent to the 120 cloth and the nexus they replaced, which resulted in reduced man-hours for layup. The additional ply of peel material over the hat stiffener improved the crown thickness per ply in the laminate. The barrier film normally applied in strips between the hat stiffeners and the underside of the caul was replaced by a continuous sheet of barrier film allowed to drape between hat stiffeners. Conventional cure stacking is shown in figure 22.

Data from tag-end test results showed the resin content to be below the minimum requirements at the root end. Therefore, the peel ply was removed from the root-end bleeder stacking in this specific area of the hat cover raising the resin content; figure 23 shows this change in bleeder stacking.

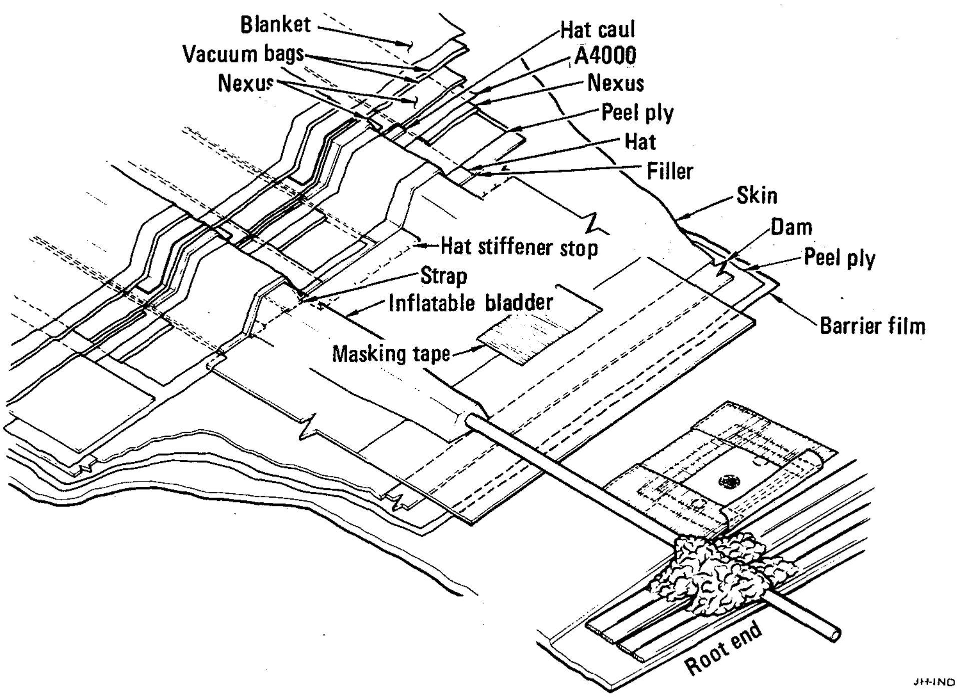
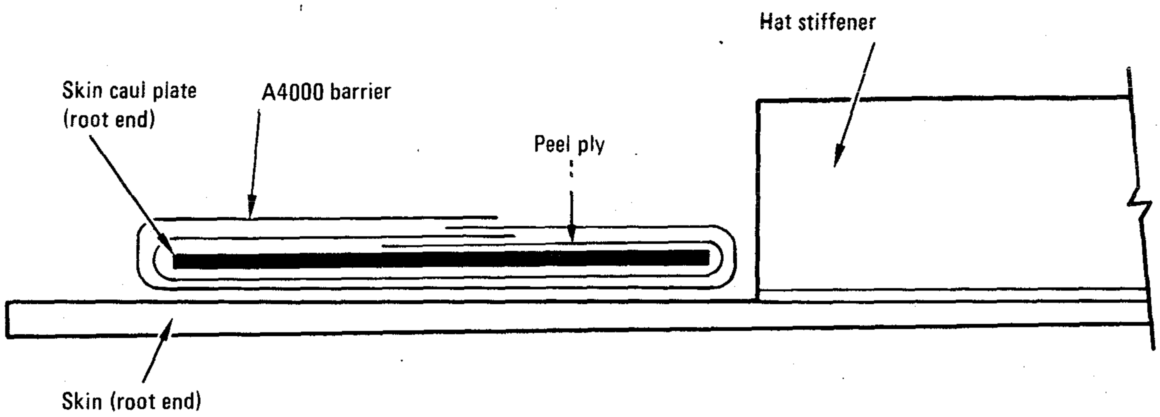


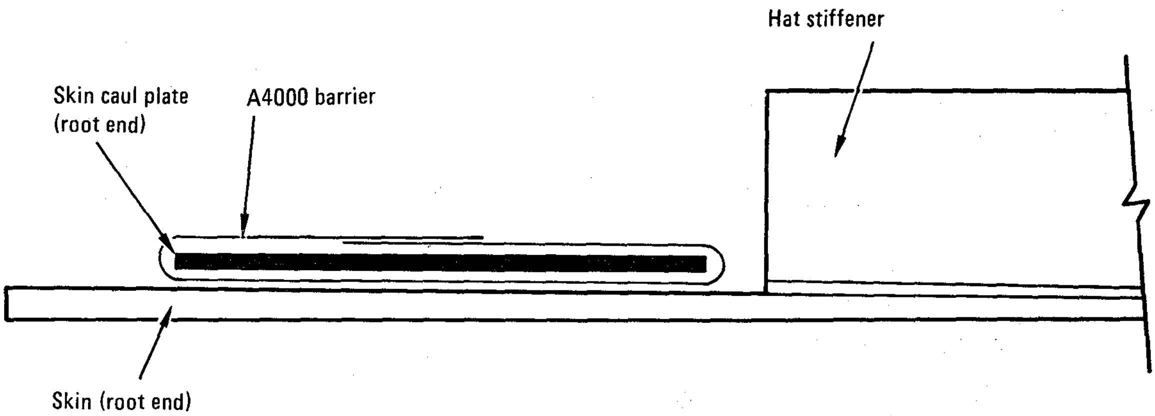
Figure 22. - Typical hat/skin assembly cure stacking arrangement.

2.2.4 Process improvements

2.2.4.1 Process procedure changes: Recurring problems with nylon vacuum bag pinholes and bag ruptures due to bridging prompted consideration of methods to safeguard against these failures. The use of a double-bag techinque was selected as the most appropriate approach to the problem. In this technique, the first vacuum bag was installed after bleeder stacking was completed by the normal practice used on previous cover assemblies. This operation was followed by application of two plies of nexus breather material over the entire surface of the first vacuum bag. This was followed by installation of the second vacuum bag, which was sealed beyond the periphery of the first bag. This was followed by the installation of four vacuum fittings consisting of a vacuum plate beneath the second bag with a pipe extending from the vacuum plate upward through the vacuum bag. The pipe was sealed to the outside surface of the vacuum bag. Individual vacuum lines were hooked up to quick-disconnect fittings at the end of each pipe. In this manner, vacuum pressure was applied to the second bag by vacuum lines independent of the lines used to apply vacuum to the first bag. The double-bag technique presented a fail-safe feature in that, if one bag malfunctioned, the remaining bag was available to permit completion of the cure. During cure, both bags were vacuum-leak checked and checked again at 345 kPa (50 psi) and at 586 kPa (85 psi) which readily satisfied all leak rate requirements. The entire cure cycle was completed without any indication of bag failure or loss of pressure.



S/N 0003 RH hat/cover



S/N 0003 LH hat/cover

Figure 23. Bleeder stacking.

2.2.4.2 Changes in layup sequence: An additional debulk cycle was added to the sequence of operations during layup. After the hat stiffeners, fillers, hat caul plates, locators, and spacers were properly positioned the layout was debulked at room temperature and vacuum pressure for a minimum of 1 hour. The vacuum bag was then removed and gaps between the locator plates and the hat stiffener caul plates were shimmed. These gaps are believed to be caused by compaction of the skin to the contour of the MBF and by slight displacement of the hat stiffeners during the debulk cycle. Appropriate shimming of the gaps was completed before finishing bleeder and breather stacking, double bagging, and cure.

2.2.4.3 Hat/cover trimming operations: On the second right hand cover assembly (S/N 0002R) a decision was made to investigate the elimination of end-of-hat trimming operations after the assembly was cured, thereby eliminating some 30 cutting operations on the assembly. This was accomplished by pretrimming the hats to net prior to laying them up on the inner surface of the cover. The hats were first laid up on their individual layup blocks, then the proper assembly trim templates were placed over the hats. The ends and edges were then cut by hand using a Stanley knife.

After trimming, the hat layups were transferred into the hat caul plates of the curing fixture (MBF). The caul plates, in turn, located the hats in relation to the cover reference lines. The end trim resulting from this method tended to produce sharp edges of resin buildup at the ends of the hats. It was found, however, that these could be easily and quickly removed by sanding with a small sanding disk mounted in a high-speed hand motor.

2.2.4.4 Reinforcing strap layup and installation: Installation of the hat stiffener reinforcing strap was a very time-consuming operation and dimensional control of each leg of the clips was very difficult to maintain. During layup of the hat stiffeners for the R/H S/N 0002 hat/cover assembly, preformed reinforcing straps, (B-staged at 381K ±8K (225°F ±15°F) for 30 to 40 minutes) were used. It was noted that man-hours required for installation was substantially reduced and the desired dimensional control was attained. Figure 24 illustrates the type of tooling used during preforming of the reinforcing straps. The mahogany layup tool is covered with a ply of A-4000 barrier film, and the reinforcement plies are manually formed in place. These layups are placed on a flat plate, vacuum bagged, and then B-staged in an oven. The reinforcing straps are carefully wrapped and stored under refrigerated conditions while awaiting installation into the hat stiffeners.

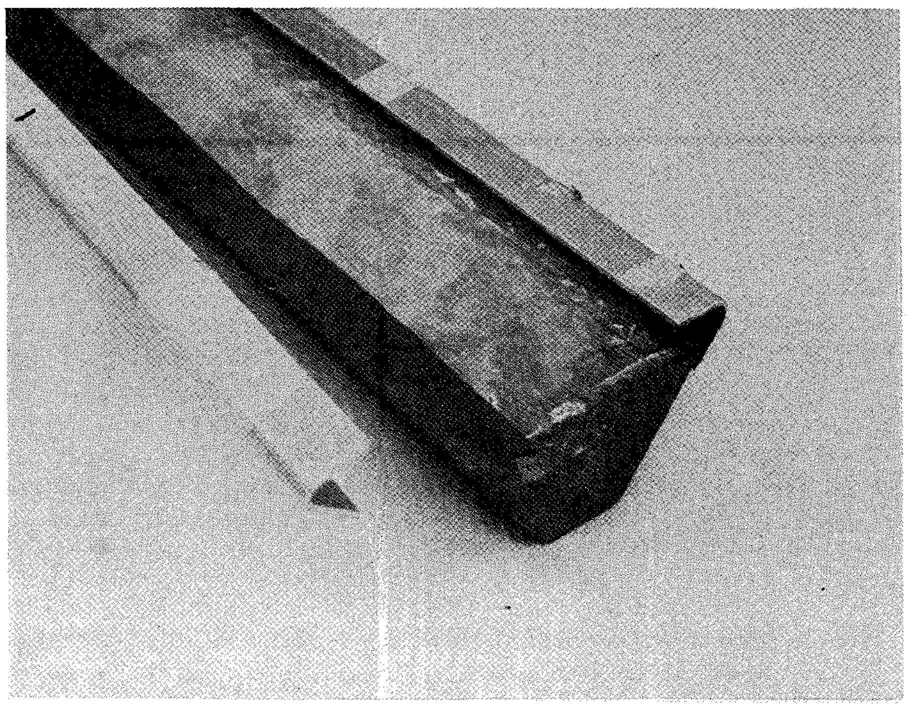


Figure 24. - Tooling for pre-formed reinforcing straps.

3. SPAR MANUFACTURING DEVELOPMENT

3.1 Tool Engineering Concept

3.1.1 Tool design features.- Elastomeric tooling for the L-1011 ACVF spars is made up of a closed mold concept with internal, heat-expandable silicone rubber mandrels.

The tool is illustrated in figure 25. Its base is essentially flat with all components of the tool and the composite spar assembled on this base in the proper sequence. Following assembly of these details, the tool cover, containing a cavity properly sized to contain the internal tool and part components, was lowered onto the base. The cover and base were accurately indexed by a novel key and slot arrangement, integral with the base and cover, respectively. A vacuum and autoclave pressure, in combination with heat-expandable rubber, apply the proper pressure to the spar during cure.

The major elements of the tool, shown in figure 25, consist of the cover, base plate, cap rails, rubber mandrels, and island blocks. Figure 26 shows the island blocks located in the tool. Appropriately spaced bleed-holes in the cap rails and island blocks provide paths between the laminate and the bleed cavities where excess resin was collected.

3.1.2 Tool manufacturing process. - Fabrication of the spar tools for the L-1011 composite vertical fin began in 1976. Heavy steel billets 17.8 cm by 81.3 cm by 7.62 m (7 in. by 32 in. by 25 ft) and plates 5.1 cm by 81.3 cm by 7.62 m (2 in by 32 in by 25 ft) were procured. Two spar tools were

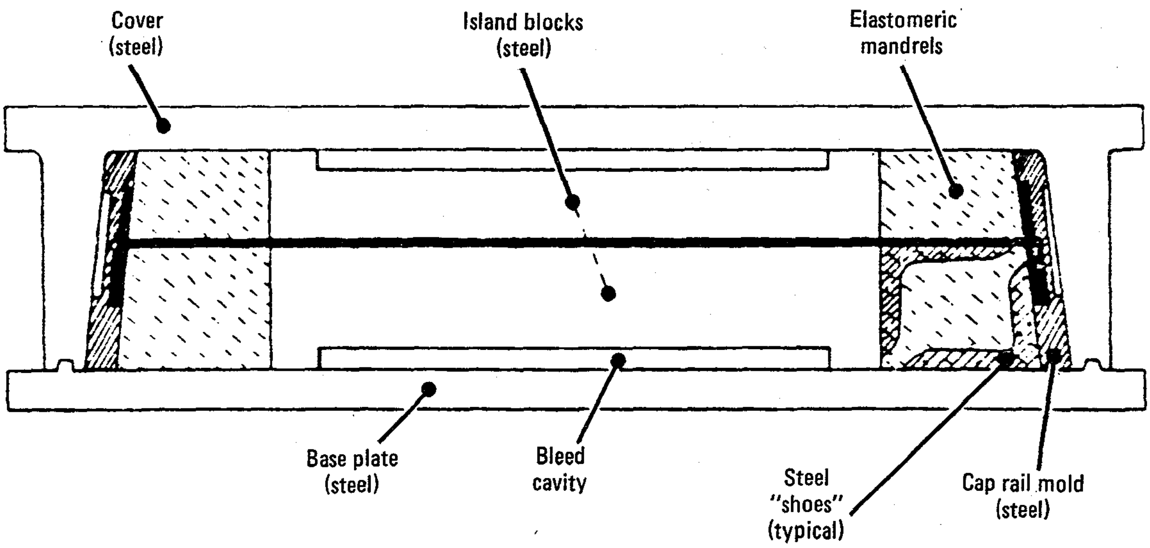


Figure 25. - Diagram of composite spar molding fixture.

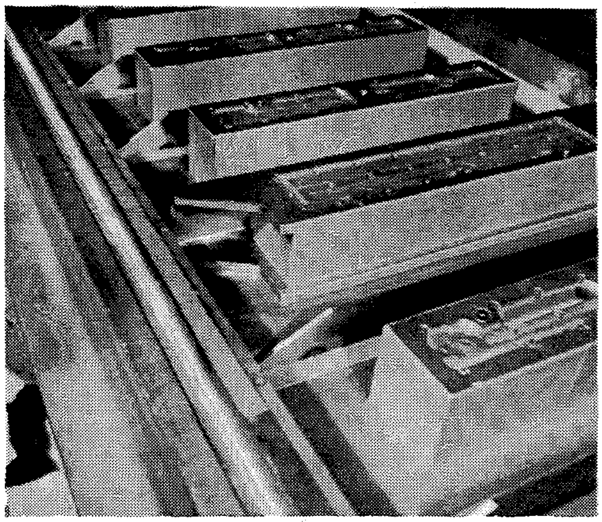


Figure 26. - Island blocks in front spar tool.

made - one for the front spar and one for the rear spar. The covers and base plates for the front and rear spars were machined from heavy steel billets, and stress relieved. Figure 27 shows a cross section of the rear spar cover and base plate.

Internal island blocks, spar cap rails, and steel shoes were precision machined to mold the critical interfaces on the spar caps, rib attaching angles, and spar web. Cavities were machined in the island blocks for reinforcing rings around access holes and for reinforcing pads on the spar web. After the internal parts of the tool were machined, they were assembled inside the tool using a dummy spar and wax. Wax was used to provide the offsets between the cured spar and the uncured spar and the thermal expansions of the rubber. Figure 25 shows a cross section of the spar molding tool.

The spar was molded using a modified thermal elastomeric process in a steel matched-die tool. The tool base, cover, island blocks, and cap rails were precision-machined steel. Thermal expansion of the cast elastomeric mandrels provided pressure against the inside of the cap flanges. Steel shoes, or channels, backed by cast elastomeric mandrels, provided pressure against the inside of the cap flanges. Steel shoes, backed by cast elastomeric mandrels, were also used at rib locations to ensure proper location, flatness, and angularity of the molded attachments. Cast elastomeric mandrels surrounded each of the steel island blocks to provide pressure against intermediate web stiffeners.

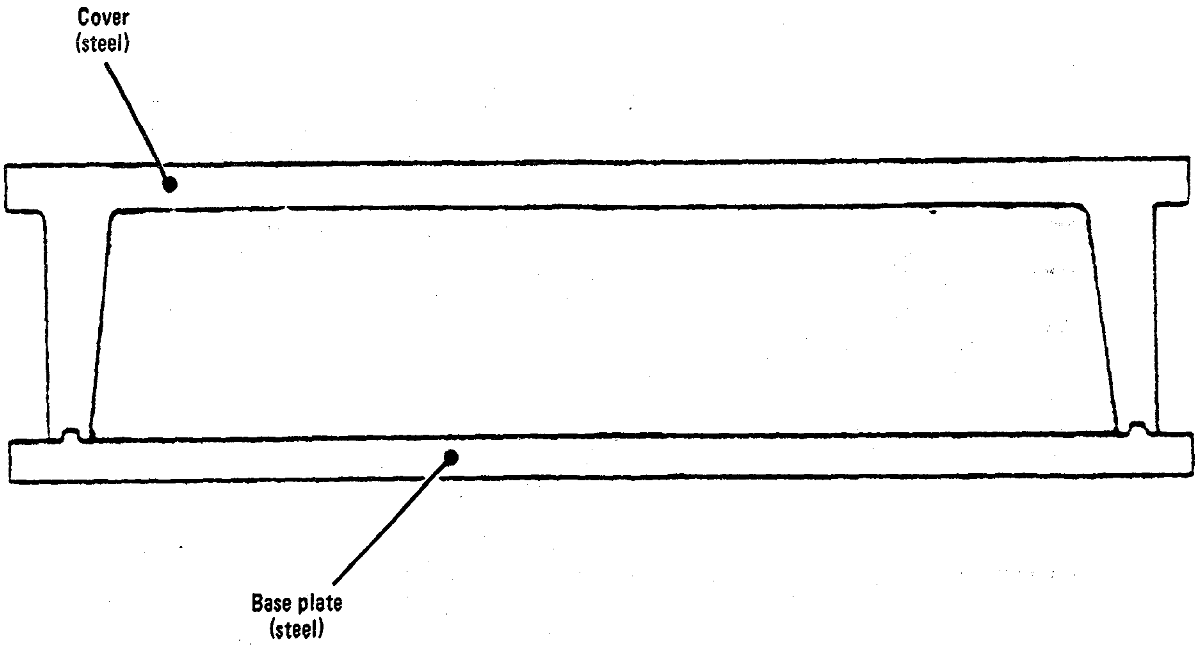


Figure 27. - Cross section of cover and base plate for rear spar near root of spar.

The elastomeric mandrels were cast with a calculated set-back (i.e., slightly undersized), so that the uncompacted prepreg could be loaded into the tool. As the mandrels expanded during tool heatup, the prepeg was compacted, and excess resin and volatiles were forced out of the laminate. The expanded mandrels maintained sufficient pressure while the resin gels and cures, to achieve proper dimensions of the finished part. Bleed reservoirs were provided within the tool on the back sides of the island blocks and cap rails. Volatiles and resin were bled through small-diameter holes drilled in the island blocks.

Steel tooling was selected to minimize tool expansion and contraction during heating and cooling. An important side effect of using steel in conjunction with elastomer is that the internal steel details allow a faster heat transfer than would be achieved via the less conductive elastomer alone.

Steel elements used to control and mold critical interfaces for ribs, covers, and fuselage fittings were remade to mold a T300/5208 stub spar used for testing. A dummy stub spar was made, and all steel elements of the tool were assembled and waxed prior to pouring the rubber mandrels. Wax sheets were used to provide the offsets between the cured spar and uncompacted, preimpregnated layups, and also to provide for expansion of the rubber during cure. Dapco blue rubber was poured and allowed to cure at room temperature.

The partially cured rubber mandrels were removed from the tool, as shown in figure 28, and cured in the oven. When all parts of the tool were complete, a fiberglass spar web and cap were laid up and cured to verify the tool.

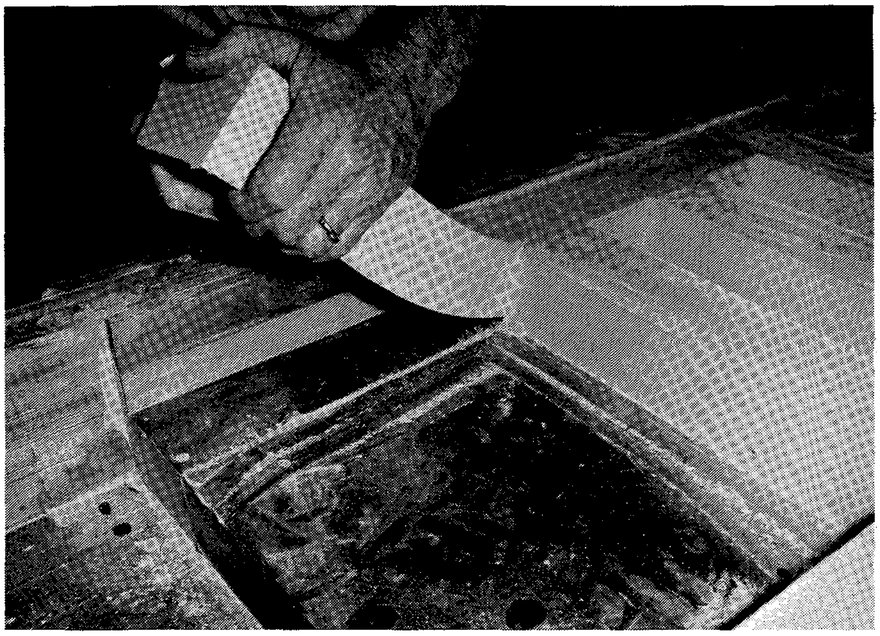


Figure 28. - Removal of rubber from tool.

A number of experiments were made during the cure of the stub rear spar. Simple steel pads were tried for molding the rib-to-cap interface surfaces on the web side of the spar caps. These pads proved successful and reduced the mark-off.

Variations of the steel angles used to mold the interface surfaces for the spar to fuselage joint were also tried. Critical tool elements that were proven during the fabrication of the two stub rear spars were incorporated in the full-size front and rear spar tools.

The T300/5208 resin used to mold stub-rear spars was very sensitive to pressure, and extreme care was taken to hold pressure off the tool until approximately 4 hours after start of the cure cycle, when the 5208 resin viscosity began to increase. The T300/5208 resin used to mold the spars was nominal 41 ±3 percent uncured resin prepreg. After the two stub-rear spars were completed, tool making proceeded for both the front and rear full-size spar tools.

Internal metal details for the front spar were made first. The rails used to mold the spar caps were machined on numerically controlled equipment. After machining the rails, fiberglass spar caps were laid up and cured. These fiberglass spar caps were used to check the rail and to provide a dummy part for casting rubber mandrels. Island blocks and steel fittings for molding the rib attaching angles were precision machined and prefitted in the front spar as shown in figure 26. Next, the large island blocks, used on the flat side of the spar, were fitted inside the cover of the front spar tool, and a dummy web was fitted to the island blocks. The dummy web gave the set back required for the production of a composite part.

After the metal details were completed, preparations were made for pouring the rubber mandrels. The dummy spar was made to the finished or cured thickness of the spar. Offsets were provided to allow for the changes of thickness of the spar during compaction and cure and for expansion of the rubber mandrels. Wax sheets were used to provide the calculated offsets between metal elements of the tool, the rubber mandrels and the cured spar.

After waxing, all elements of the tool were assembled, clamped and prepared to receive the liquid rubber mix. One end of the tool was raised to allow the air to escape, while rubber was forced in at the lowered end of the tool on the flat side of the spar. Next, the tool was leveled, and rubber was gravity fed to fill the openings between the stiffeners, as shown in figure 29. The rubber was allowed to cure at room temperature and the clamps were removed. After initial cure at room temperature, the island blocks were removed. The partially cured rubber was removed from the tool and cured in the oven.

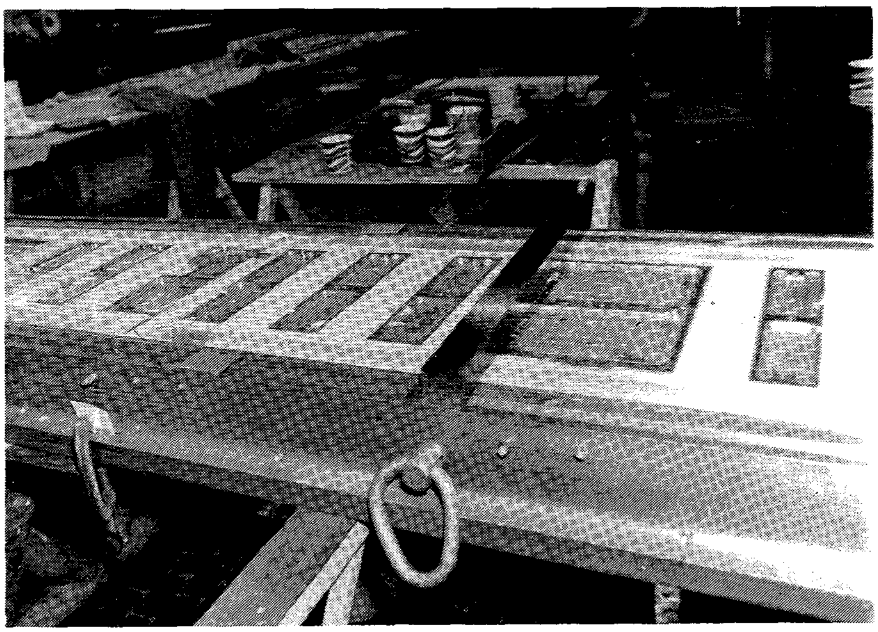


Figure 29. - Gravity leveled rubber around island blocks completed.

Before fabrication of the first full-size front spar, a fiberglass tool try part was made to verify the tool. The fiberglass spar was measured, and it was verified that the tool would mold the spar to the required engineering dimensions.

3.1.3 Tool proving. - Originally, the first spar made in the front spar tool was to be destruction tested as a tool-proving article, but, when inspected, this spar was found to conform with the engineering and process requirements, and a decision was made to wait until all the spars were fabricated and then select the spar with the worst inspection record for destruction testing. Three more front and three rear spars were fabricated, and these three ship sets of spars were also found acceptable for use in the assembly of the fin box. The seven spars are shown in figure 30.

Selection of a spar for destruction testing was difficult, because all spars were found to conform to the engineering requirements. Some minor discrepancies existed in all spars, but none of the discrepancies were sufficient for rejection of a spar. The tool try first spar was thicker in the web edges than on Spars 1, 2, and 3. This overthickness did not present any fit-up problems, but it was corrected. Correcting the web edge thickness caused areas in the aft flange of Spar No. 1 to be near or below the minimum thickness tolerances in local areas. Another adjustment was made to the tool, and the second and third spars were practically free of discrepancies.

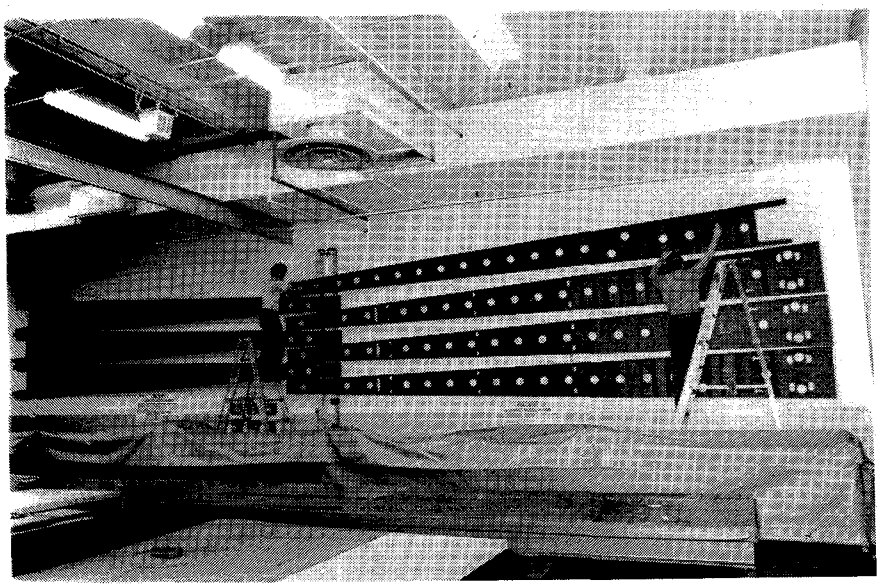


Figure 30. - L-1011 ACVF spars for 3 ships and 1 tool try.

Process control specimens cut out of the access holes showed front Spar No. 1 had the lowest resin content of the seven full-size spars. Resin contents for the full size spars are compared below:

Spar	Average Resin Content - % Wt. (Specification, 26 to 34%)
Front Spar Tool Try	29.2
Front Spar No. 1 (Destr. Test)	26.3
Front Spar No. 2	28.5
Front Spar No. 3	29.8
Rear Spar No. 1	29.4
Rear Spar No. 2	29.0
Rear Spar No. 3	31.3

Because of the thinner areas in the cap flange and lower resin content, front Spar No. 1 was selected for destruction testing. The coupons cut from this spar demonstrated that the process and tools were satisfactory.

3.2 Manufacturing Process Description

The spars were fabricated with T300/5208 graphite-epoxy, unidirectional tape. The Thornel 300 fiber had an areal mass of 144 grams per square meter, and it was preimpregnated with an uncured, NARMCO 5208 resin content of 41 $\pm$3 percent by weight. The nominal cured thickness was 0.0127 mm (0.005 in.) per ply.

The numerous plies used to build up the spars were laminated, trimmed, and assembled into caps, rib angles, stiffeners and webs, and loaded into the spar tools for initial cure in the autoclave. Quality of the spars was ensured by planned inspections during the preparation of the material, trimming, layup of the plies, assembly of the spar, loading in the tool, and during cure. Process control specimens were taken from the spar webs and tag ends to verify physical and mechanical properties.

3.2.1 Broadgoods layup and pattern cutting.- Procedures for laying up broadgoods and cutting out the various patterns were similar for both the front and rear spars. These layups are easily adaptable to automation and computer-derived ply patterns. The front spar has six plies of ±45-degree ply orientation on the outer faces of the web. Both faces can be laid up as a 6-ply-wide laminated sheet, and a pattern can be used to cut the two outer faces of the spar web. Similar groups of ply orientations are used for the remaining elements of the spar webs and for all of the other elements used to assemble the spars.

Layup of broadgoods began with removal of the T300/5208 preimpregnated tape from the freezer and cutting the 30.5 cm (12 in.) wide tape to the required length. Each precut length of tape was laid in place. In some cases, the stiff tape required the application of heat to force the tape to lie flat.

Spar elements to be cut from each layup of the broadgoods were nested on a computer printout. Reduced size computer printout showed the various elements arranged to be cut with a minimum amount of waste material. Full-size computer-printed pattern was laid on top of the broadgoods. The elements of the spar were cut to the full-size pattern laid on top of the broadgoods, and the smaller elements were trimmed to metal templates. The procedure is adaptable to an automated, computer-controlled cutter. The large number of small elements used to laminate and assemble the spar can be cut from the broadgoods with less than 5 percent of the material wasted. Elements cut out of the broadgoods were identified, fitted, and returned to the freezer to be recalled as needed. Elements of the stiffeners were preassembled in wooden blocks with the web flanges preformed prior to stowing in the freezers.

3.2.2 Assembly of composite spar details. - Assembly of the composite spar in the spar mold tool was a relatively simple operation. The tool was prepared to receive the various preplied elements of the spar. These spar elements were put together in kits and partially assembled. First, the preassembled stiffeners and rib attaching angles were placed in their locations, as shown in figure 31. Next the spar web was located on top of the stiffeners, using tooling pins.

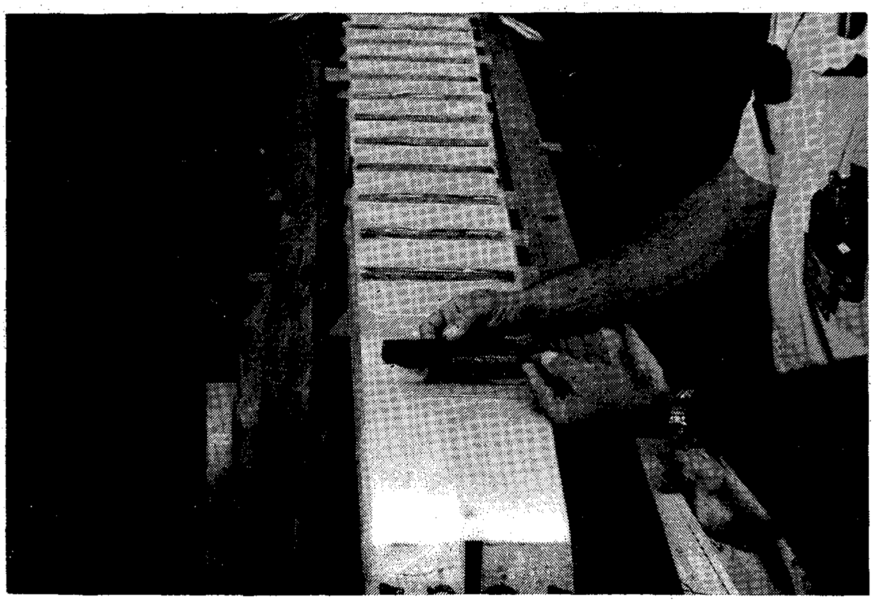


Figure 31. - Loading stiffeners in front spar tool.

After the web was in place, a sheet of Armalon was placed on top of the web to provide a breather between the web and island blocks. The 6 plies of ±45 degrees on the outer faces of the spar web extend along the lengthwise edges of the spar. These plies are folded and trimmed to form the inner flanges of the spar caps. The cap rail molds containing the remaining cap plies were then placed in position completing the assembly. Clamps are used to compact the spar cap rails as shown in figure 32, prior to placing the cover on the assembled spar and tool. The cover is placed on top of the assembled spar as shown in figure 33, and the large number of thermocouples are carefully routed through the vacuum bag. Figure 34 shows the vacuum bag being placed on the tool and the vacuum bag check before going into the autoclave.

A long cure cycle was required to cure the spar because of the mass of the tool and the time required for heat to be transferred through the steel and rubber tooling to the graphite-epoxy spar. The cure cycle was a two-part cycle. The initial cure required approximately 17 hours in the autoclave. Approximately 9 hours were required for the spar to reach the 394K (250°F) initial cure with full pressurization. During the heat-up and cure, thermocouples, vacuum, autoclave, pressure, and temperature were recorded every 10 minutes, and adjustments were made to maintain uniform temperatures inside the tool. The spar was cured approximately 6 hours at 394K (250°F) and 965 kPa (140 psi) followed by a cool-down and removal from the autoclave. After the tool cooled to a temperature at which it could be handled, the vacuum bag was removed, and the tool disassembled.

Figure 32. - Final compaction of all internal parts of front spar tool prior to cover.

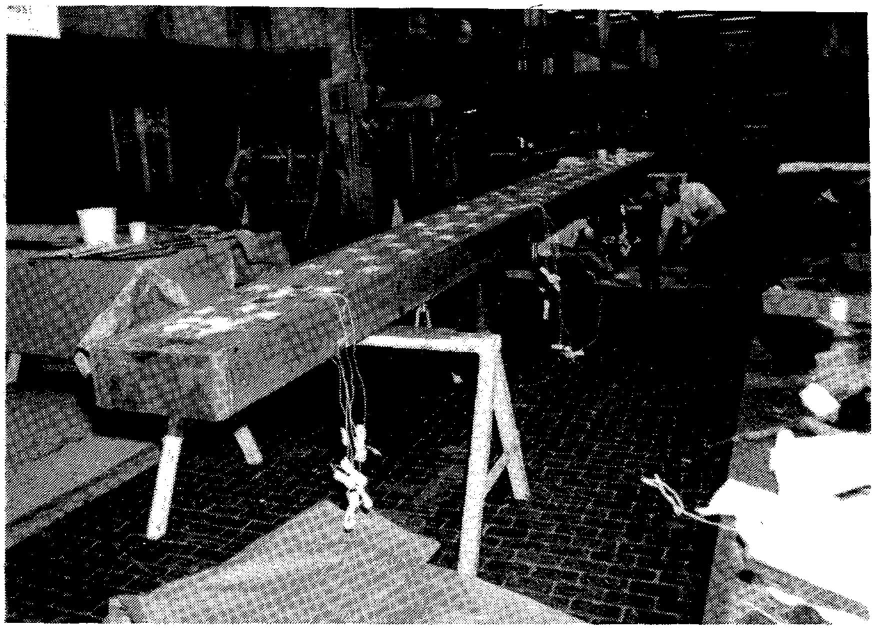


Figure 33. - Cover placed on front spar and prepared for bagging.

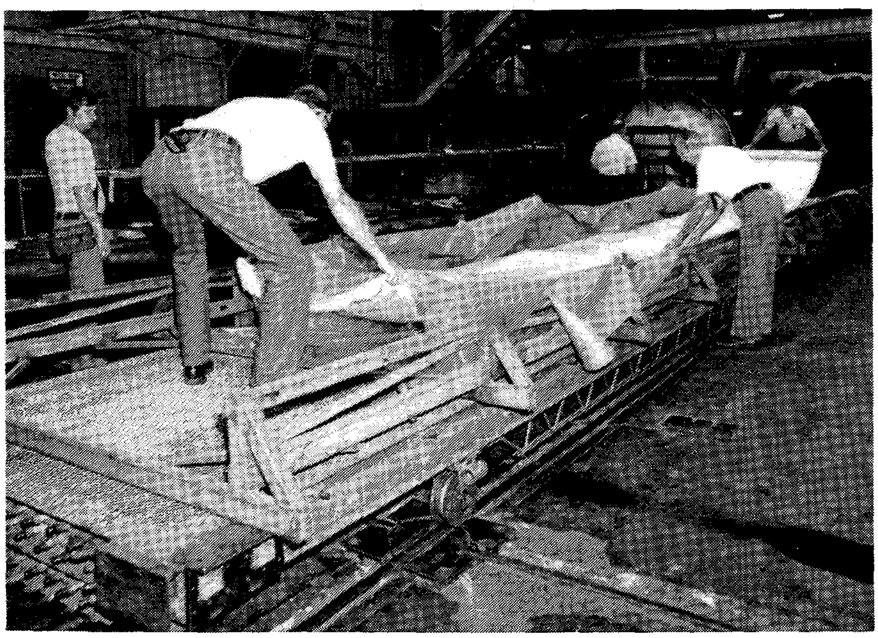


Figure 34. - Preparing bag for front spar in tool at autoclave.

The first step in disassembly of the tool was to lift the base plate off and remove the steel island blocks and rubber mandrels from the stiffener side of the spar as shown in figure 35.

Next, the steel rails used to mold the spar caps were pried loose, and the baseplate was replaced. This left the stiffeners free, and the spar was supported on the stiffener side by the rail caps when the tool was turned upside down. After turning, the cover was lifted off.

The steel island blocks, cap rails, and long rubber mandrels on the flat side of the web were removed and the spar was lifted from the tool. While the spar was being cleaned and prepared for post cure, the tool was cleaned and reassembled for the next spar. Bled resin was cleaned out of the cover and base plate with relative ease because of the Frekote. Resin that had accumulated in the bleed holes of the steel island blocks was drilled out. Residual resin on the rubber mandrels was removed and a free coat was applied. Frekote was applied to all parts of the tool prior to reassembly for the next spar.

Cleanup of the spar prior to post cure required the residual strips of Armalon to be removed. The small amount of resin flash around the edges of the spar caps and stiffeners was easily removed by light sanding. Figures 36 and 37 show the spar after removal from the tool prior to postcure.

Postcure of the spar was accomplished by placing the spar on a flat grid to allow good air circulation and curing it in the oven at 450K (350°F), as

Figure 35. - Base plate lifted off and removal of rubber blocks and lower steel block.

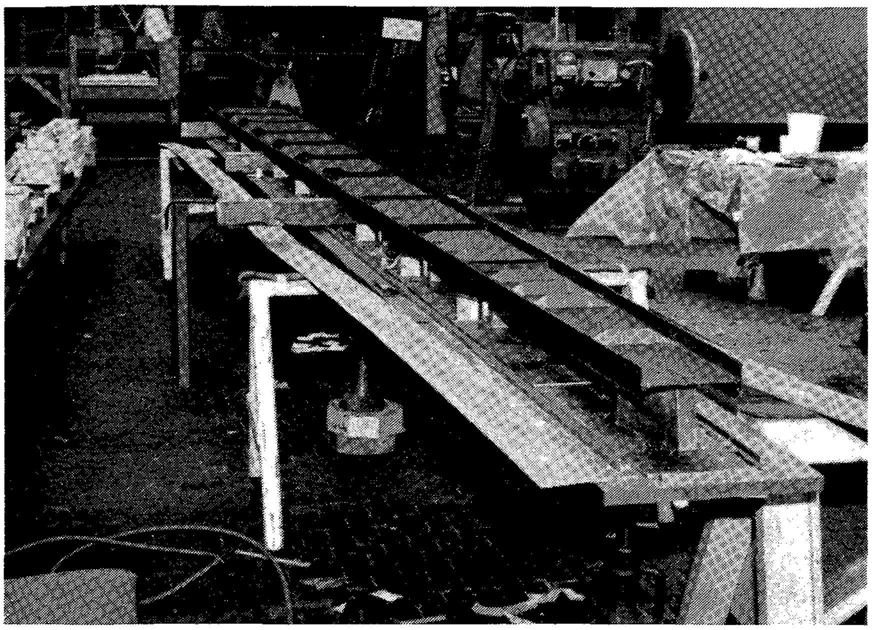


Figure 36. - Spar as removed from tool viewed from upper end.

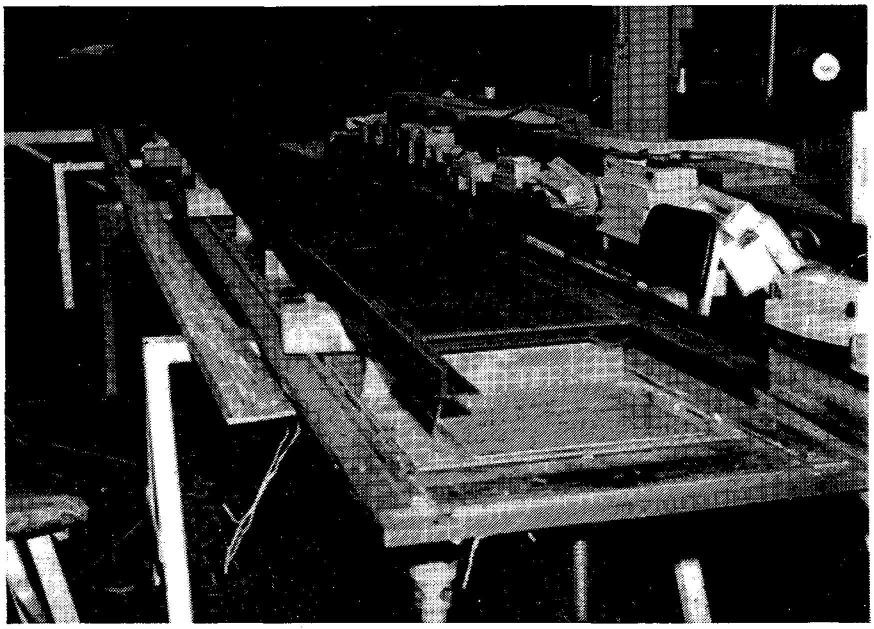


Figure 37. - Spar as removed from tool showing forward side.

shown in figure 38. During postcure, the spar was unrestrained, and no measurable change in dimensions or warpage could be detected.

After postcure, the access holes were machined in the spars.

A center hole for the disc cutter was drilled, and a 10.2 cm (4 in.) diameter diamond cutter was used to cut the discs from the spar web. The D-holes in the rear spar molded in place and did not require additional machining.

The discs were used for process control specimens, as shown in figure 39. Five types of specimens were cut from the disc. These were as follows:

P - porosity, polish, and microscopic check
Sp. Gr. & RC - Specify Gravity and Resin Content
Flex - Flexural Bending
Comp. - Compression Strip
SBS - Short Beam Shear

3.2.3 Spar cure cycle. - A two-step process was developed and verified for curing the spars. This process is compatible with the modified elastomeric tooling design and uses an autoclave cure, followed by an oven postcure. The spar is removed from the tool before postcuring.

The final spar cure cycle, including processing tolerances, is illustrated in figure 40.

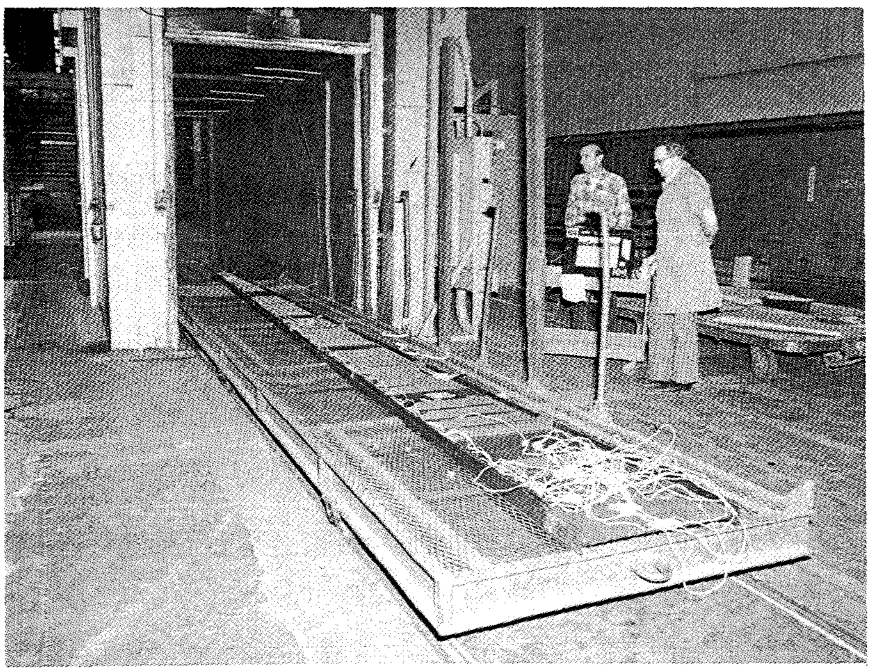


Figure 38. - Post cure of spar at 450K (350°F)

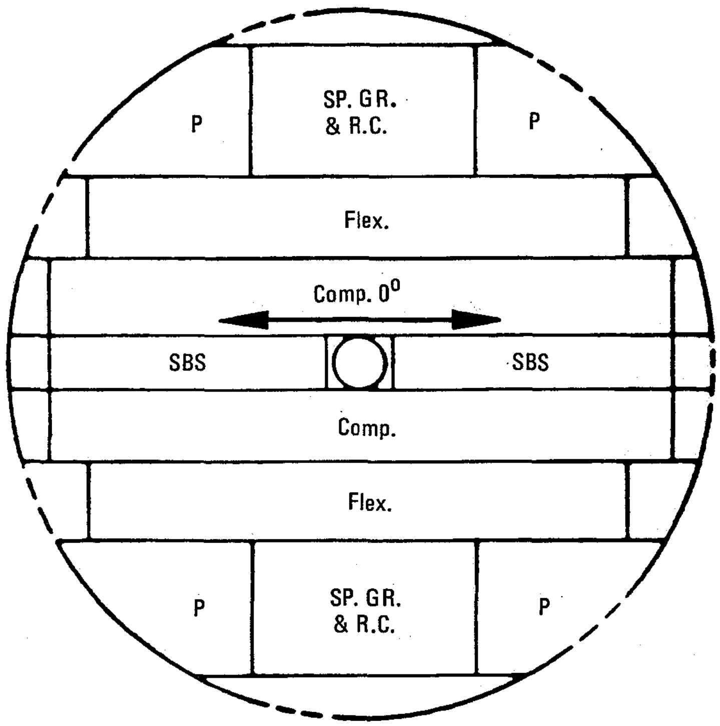


Figure 39. - Process control specimens taken from access holes.

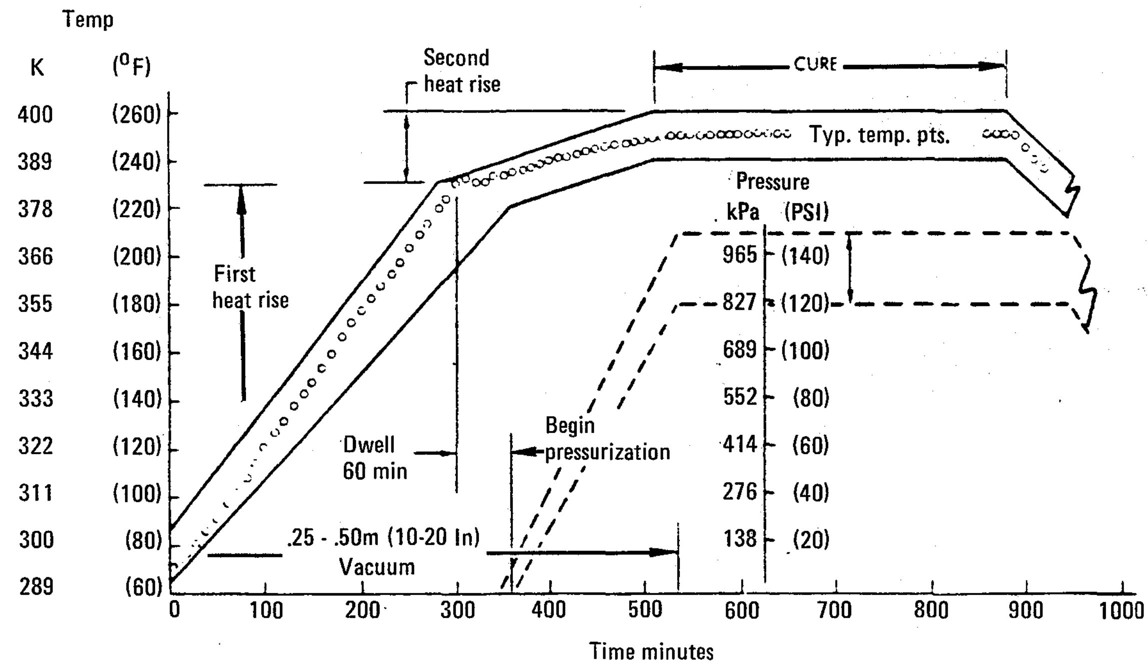


Figure 40. - Cure cycle for L-1011 ACVF spars.

The cost for the first three full-length, L-1011 ACVF spars was considerably less than the estimated production hours (T_1) made in 1977. Figure 41 shows a graphical history of the labor estimates for production of the spars and compares these estimates to the actual hours for the first three. The original estimate for fabrication of the first ship set of spares for the L-1011 ACVF was 3730 hours, and the original T_1 for production (fourth unit) in 1977 was 2600 hours. The considerably lower actuals are explained below and they are consistent with the methodology for the production estimate.

The full-length spars were fabricated concurrent with the production readiness verification test (PRVT) spars. Experience was gained by the technicians during the fabrication of the PRVT spars, and numerous manufacturing efficiencies were incorporated. Computer-graphics augmented design and manufacturing (CADAM®) was used to produce ply templates; parts were put together in kits for more than one unit, and semihard tooling and other techniques were used. These techniques were similar to those identified during the producibility studies, and analysis of the cost tracking data indicated the production costs for fabrication of the composite spars would fall very close to the 1241 hours predicted for the third unit in the producibility studies.

Assembly of the metal parts to the composite spar was performed at a much lower cost than expected. Actual time for the first spar was 207 hours

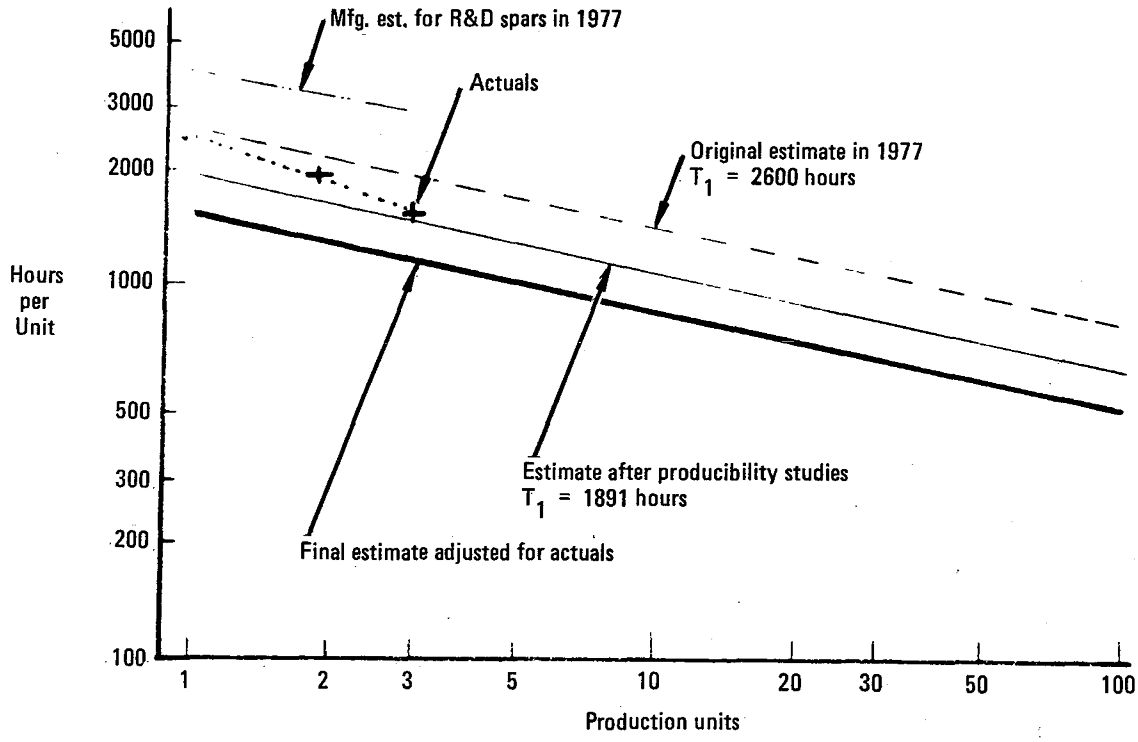


Figure 41. - Production spar estimates and actuals for first three L-1011 ACVF spars.

® A registered trademark of CADAM, Inc.

as compared to the original 684 hours for the first research and development spar and 579 hours for the production T_1. Drilling was not a major problem, and unknown assembly difficulties did not materialize. Based on actuals, T_1 for the production estimate was reduced to 200 hours. After adjusting the subassembly hours, the final production estimate shown in figure 60 has T_1 equal to 1511 hours.

The metallic and composite cost estimates were projected on the same basis to ensure comparable results. Both the composite fin and the metal fin were assumed to start from the first production unit (T4). Costs were expressed as recurring production costs and included production labor, material, quality assurance, and sustaining engineering and tooling.

4. RIB ASSEMBLY MANUFACTURING DEVELOPMENT

4.1 Tool Engineering Concept

4.1.1 Tool design features

- Interchangeability: Since the ribs are not classified as interchangeable, no consideration was necessary for interchangeability.

- Critical interfaces: Although none of the fit relationships of the rib assemblies to the box assembly required exceptionally tight control, the rib cap cutouts to cover assembly hat stiffener location was controlled to ensure interference-free assembly and installation capabilities. This was controlled by means of master tooling templates as described in the cover assembly section.

- Tool manufacturing cost: Selection of a male layout tool for the rib caps was made because it eliminated the possibility of rib cap laminate bridging across the internal radii of a female tool during cure. A matched mold would greatly reduce part fabrication time but the cost of this type of tooling for three shipsets of ribs was prohibitive.

4.1.2 Tool manufacturing process

4.1.2.1 Fabrication tooling: The rib cap layup tools were conventionally machined from solid alpase cast aluminum tooling plate with pockets on the bottom side of the tool to reduce the heat-up time and provide even heat-up. Thermocouples and a vacuum system were made an integral part of the tools. To reduce the cost of machining the tools, the flange angle variances from front to rear spar ends of the tools were averaged to a mean angle due to the range of variance on either side of the mean being within standard tooling tolerances.

4.1.2.2 Assembly tooling: The truss and actuator rib assemblies were assembled on wooden frame bench type assembly fixtures. The aluminum angle locators for indexing and clamping the rib caps and webs were set to engineering master drawing photographs on metal template stock fastened to plywood backing. A typical fixture is shown in figure 42.

4.1.3 Tool proving problems and solutions

4.1.3.1 Fabrication tool proving: One of the most time consuming and costly problems encountered during the tool-proving program was vacuum leaks. Aside from the normal recurring causes for vacuum loss, which include faulty autoclave operation and bag failure, integral plumbing, vacuum fittings, and thread sealant appeared to be the problem areas. In one instance, the aluminum layup tool material was found to be porous.

- Integral plumbing: to avoid leakage, integral plumbing was removed from the rib layup tools.

- Vacuum fittings: The type of quick-disconnect vacuum fittings being used originally on the rib layup tools were found to be deficient in design. The valve stem portion of the fittings had a very short bearing surface. After some use, the bearing surface diameter became worn, allowing the valve stem to move off center of the valve seat. This caused poor seating and prevented successful leak checks.

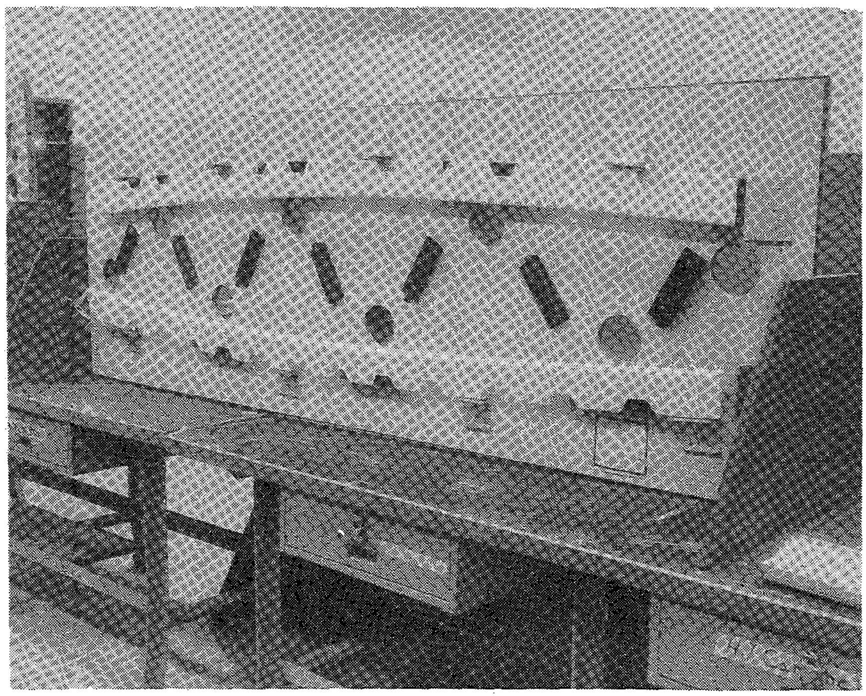


Figure 42. - Two truss rib caps held in assembly fixture.

- Thread sealant: The problem with the integral plumbing, or any type of plumbing, was providing a good seal in the threads. Various types of thread sealant were used, including Teflon tape. High-temperature silicone rubber RTV sealant was found to be the best.

- Porous tool material: An isolated case of vacuum leakage came from porous tooling material. To alleviate the immediate problem, the tool and part were envelope bagged. The supplier of the MBF raw material offered a sealant material to be sprayed on the tool to seal any porosity. The spray was not used as it may have contaminated the parts during cure.

4.1.3.2 Assembly tool proving: The rib assembly tools required no tool-proving corrections due to the straightforward sheet-metal method of fastening the components. Tempered masonite backup blocks were provided for back-drilling fastener holes from the extruded aluminum diagonals into the graphite composite rib caps and webs. In addition to backing up the composite material while drilling, the drill feed was restricted to provide controlled exit pressure at breakthrough.

4.2 Manufacturing Process Description

Figure 43 shows the cured composite ribcaps and solid web ribs (some of which are painted). The truss and actuator rib caps are all 19-ply laminates with identical ply orientations. The two actuator rib webs are 16-ply laminates. The three solid web ribs have syntactic core webs with 7-ply face sheets of graphite-epoxy, the flanges are solid 20-ply graphite-epoxy.

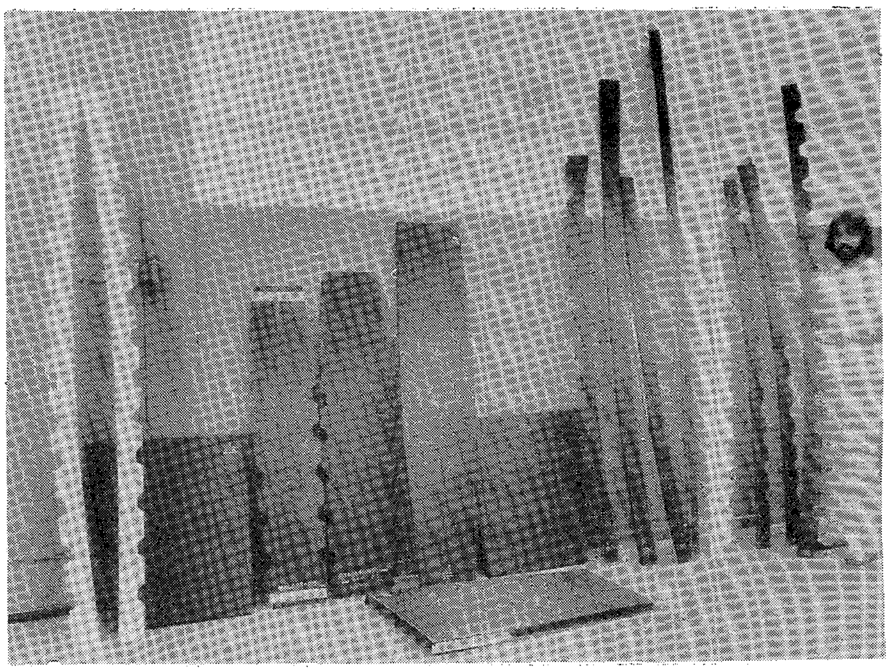


Figure 43. - Cured composite parts.

4.2.1 Rib layup and cure.- The material for the ribs is cut and preplied in three- or four-ply stacks. See figures 44 through 46. These stacks are then positioned on warmed aluminum tools that have baked-on release coat. After positioning each preplied stack, the material is vacuum debulked. At the completion of layup, the flanges are trimmed back to be 12.70 mm (0.5 in.) above the tool base plates. The laminate is then lifted off the tool and a barrier film draped over the tool. The laminate is then replaced on the tool and damming tape is butted up to its edges to minimize edge bleeding. The prepreg is then covered with Armalon (a Teflon-coated cloth) which is trimmed net to the edge dams. A 0.05 mm (0.001 in.) thick perforated barrier film is then placed over the Armalon. The perforated barrier film reduces resin bleed while allowing volatiles to escape. This film is also trimmed net to the edge dam. Next, an Armalon breather ply is placed in position. This breather extends 101.60 mm (4 in.) beyond the perimeter of the part. A 2.29 mm (0.09 in.) thick formed silicon rubber caul is then fitted over the assembly.

This caul was fabricated by molding net to the tool to produce a wrinkle-free surface. Two plies of a polyester breather cloth are draped over the caul, and a nylon vacuum bag is sealed over the assembly to a tooling plate. The resulting cured tool surface of the part is smooth and requires light sanding to remove the gloss prior to paint. The bag side surface did not require any surface preparation prior to painting. Previously, a nylon peel ply was used next to both surfaces of the laminate. This gave excellent surface finish for painting; however, the resulting resin contents were near or below the minimum of 26 percent by weight specified in the Process Bulletin. Substituting porous Armalon for the peel ply on the bag side did not sufficiently improve resin content, so the peel ply on the tool side was removed, and the resin content then moved into the middle of the specification range of 26 to 32 percent.

4.2.2 Rib bleeder/breather stacking.- The bleeder stacking for the ACVF rib components was developed to accommodate a standard autoclave cure cycle for the four configurations of rib components. Figure 47 shows the solid web bleeder stacking arrangement. The bleeder stacking arrangement for the balance of components is depicted in figure 48. The solid web stacking differs from other rib stackings to permit bleeding from both sides of the core in the solid web rib.

4.2.3 Rib cure cycle. - All composite rib components were cured in a space-heated autoclave pressurized with CO_2 using the curve cycle shown in figure 49.

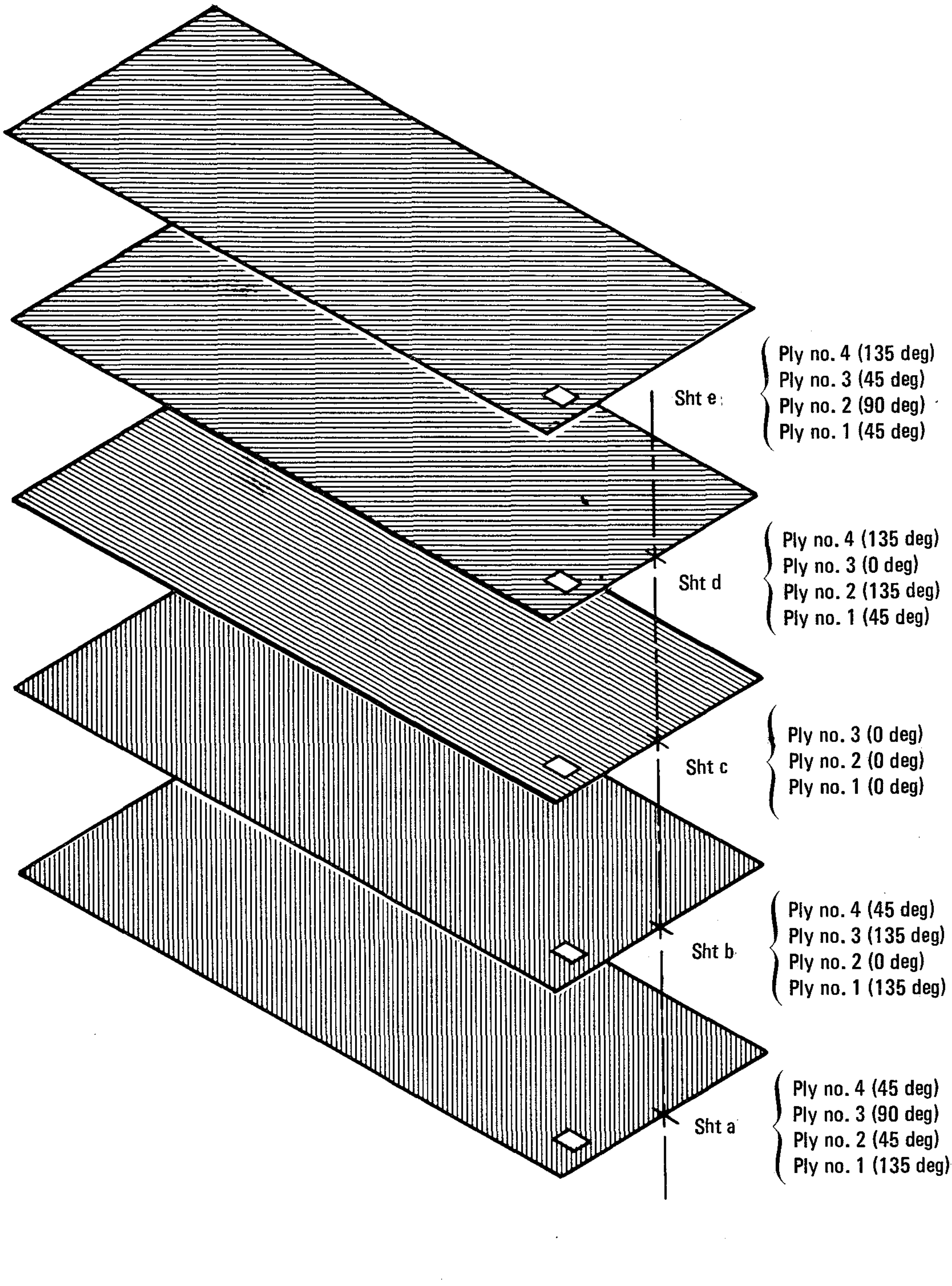


Figure 44. - Typical truss and actuator rib cap preplied stacks.

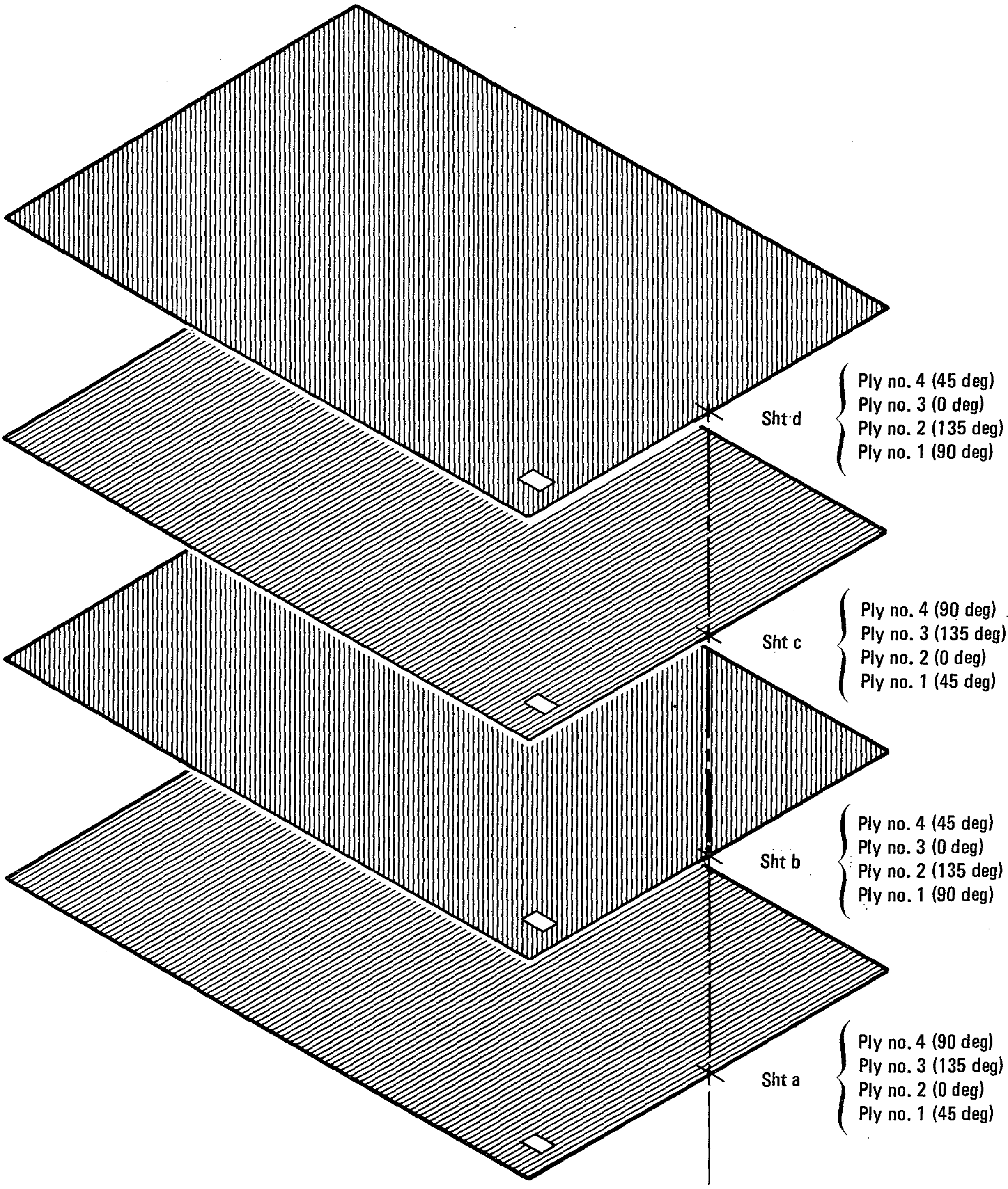


Figure 45. - Typical actuator web preplied stacks.

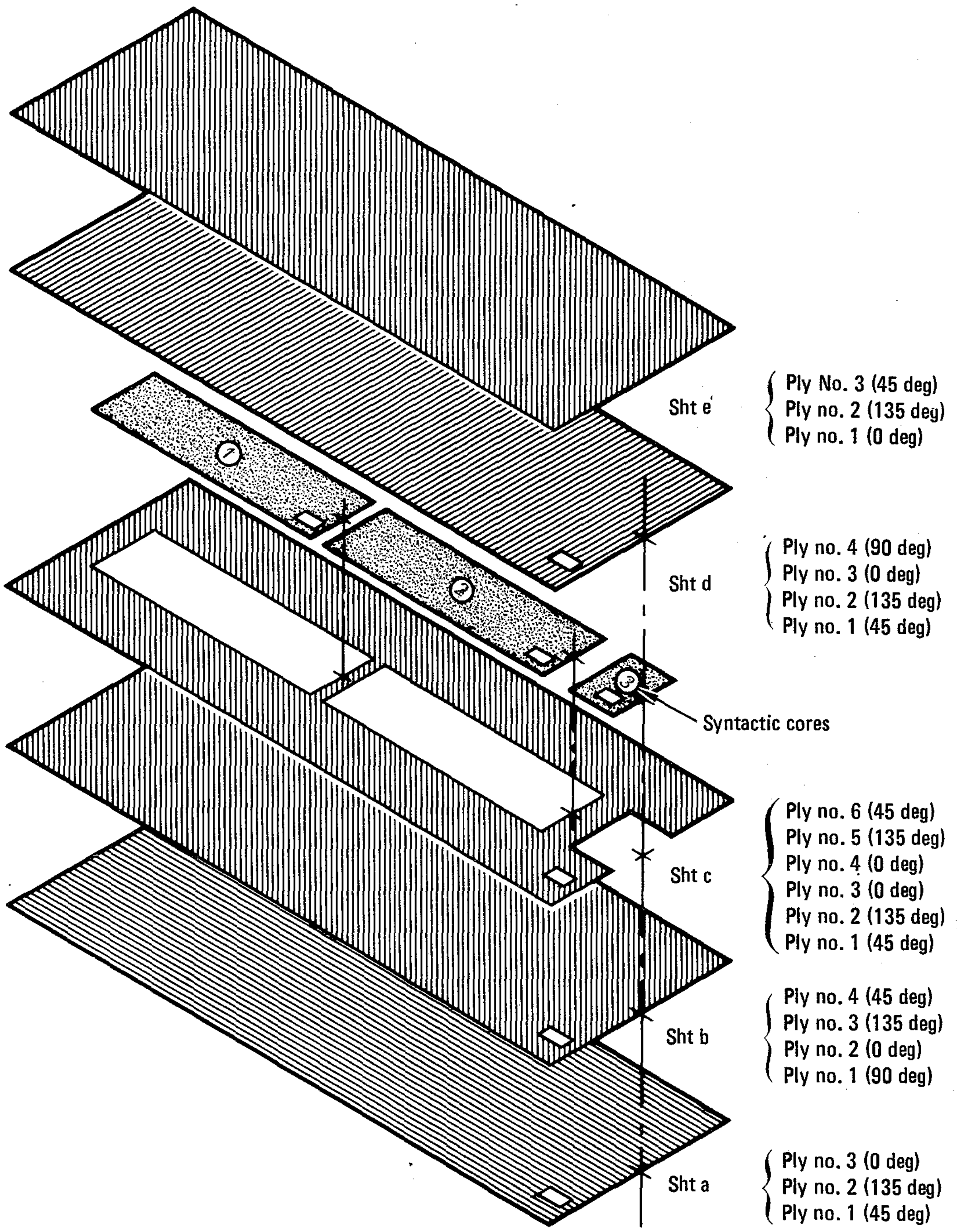
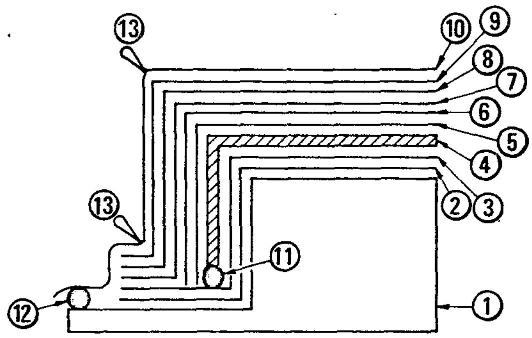
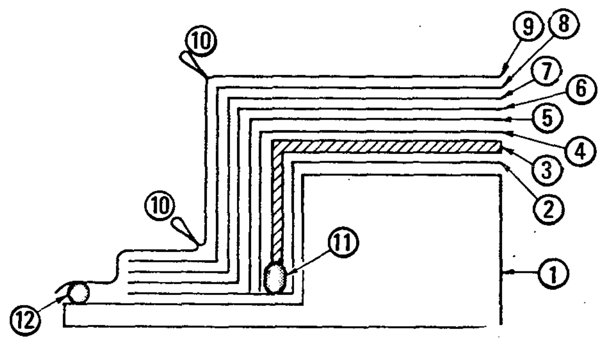


Figure 46. - Typical solid web rib preplied stacks.



① Layup tool
② Barrier film (A-4000)
③ Teflon coated cloth (Armalon)
④ Prepreg. layup
⑤ Armalon (or equivalent)
⑥ Perforated separator film (A-4000P)
⑦ Teflon coated cloth (Armalon)
⑧ Silicon rubber bag
⑨ 2 plies air weave or nexus
⑩ Vacuum bag
⑪ Air dam
⑫ Bag sealant
⑬ Dog ear

Figure 47. - Solid web rib bleeder stacking arrangement.



① Layup tool
② A-4000 barrier film
③ Prepreg. layup
④ Teflon coated cloth (Armalon)
⑤ A-4000P perforated film
⑥ Teflon coated cloth (Armalon)
⑦ Silicon rubber bag
⑧ 2 plies air weave or nexus
⑨ Nylon bag
⑩ Dog ear
⑪ Air dam
⑫ Bag sealant

Figure 48. - Actuator and truss rib bleeder stacking arrangement.

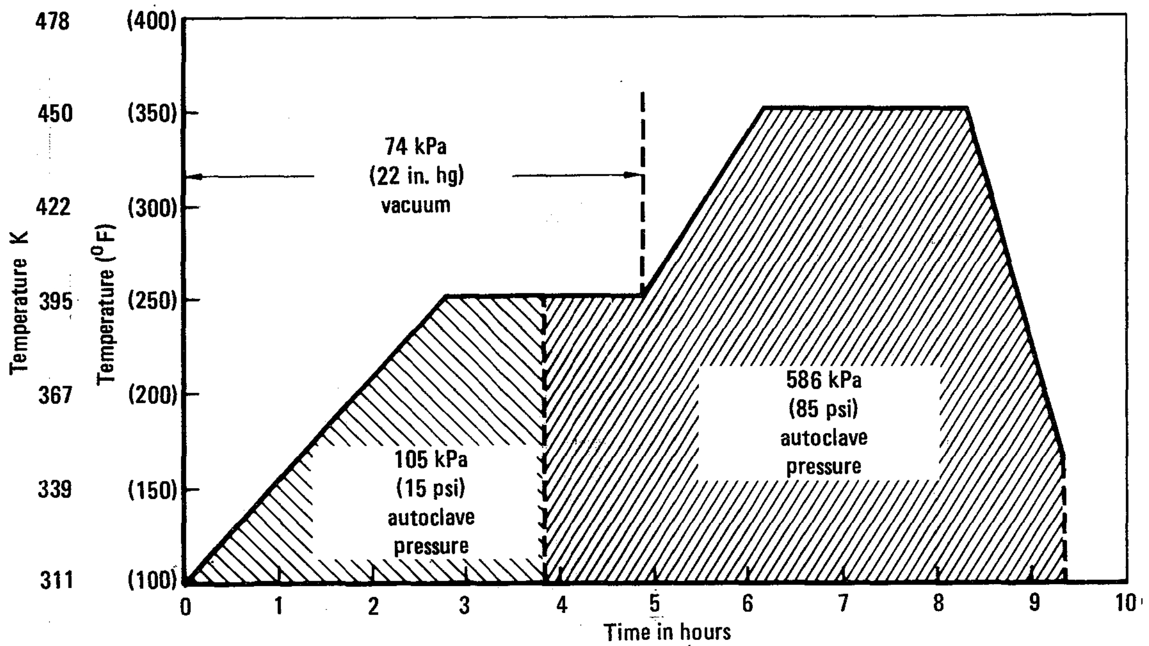


Figure 49. - Rib cure cycle.

5. FIN BOX ASSEMBLY

5.1 Fin Assembly Tool Engineering Concept

The composite fin box was assembled at the production L-1011 empennage assembly plant in Meridian, Mississippi. One of the L-1011 metallic fin assembly fixtures and other related tools were reworked for use in the assembly of the composite fin. These tools were converted back to their original condition after the composite fins were assembled.

Tool details needed for the assembly tools were fabricated at Marietta, Georgia, and shipped to Meridian. A team of Lockheed-Georgia tooling personnel went to Meridian and modified the assembly tools that were to be used for the assembly of the composite box. The two major tools altered were the fin box assembly tool shown in figure 50 and the trailing edge fixture. Other tools such as jig drill plates (JDP) and slings were altered as needed prior to the start of the assembly operations.

Coordination meetings were held between Lockheed-California and Lockheed-Georgia to resolve potential assembly interface problems that might occur due to tolerance buildups of the ribs, covers, and spars. Agreements were incorporated in the Contract Source Book, a document defining the state of assembly of parts shipped to Meridian, Mississippi.

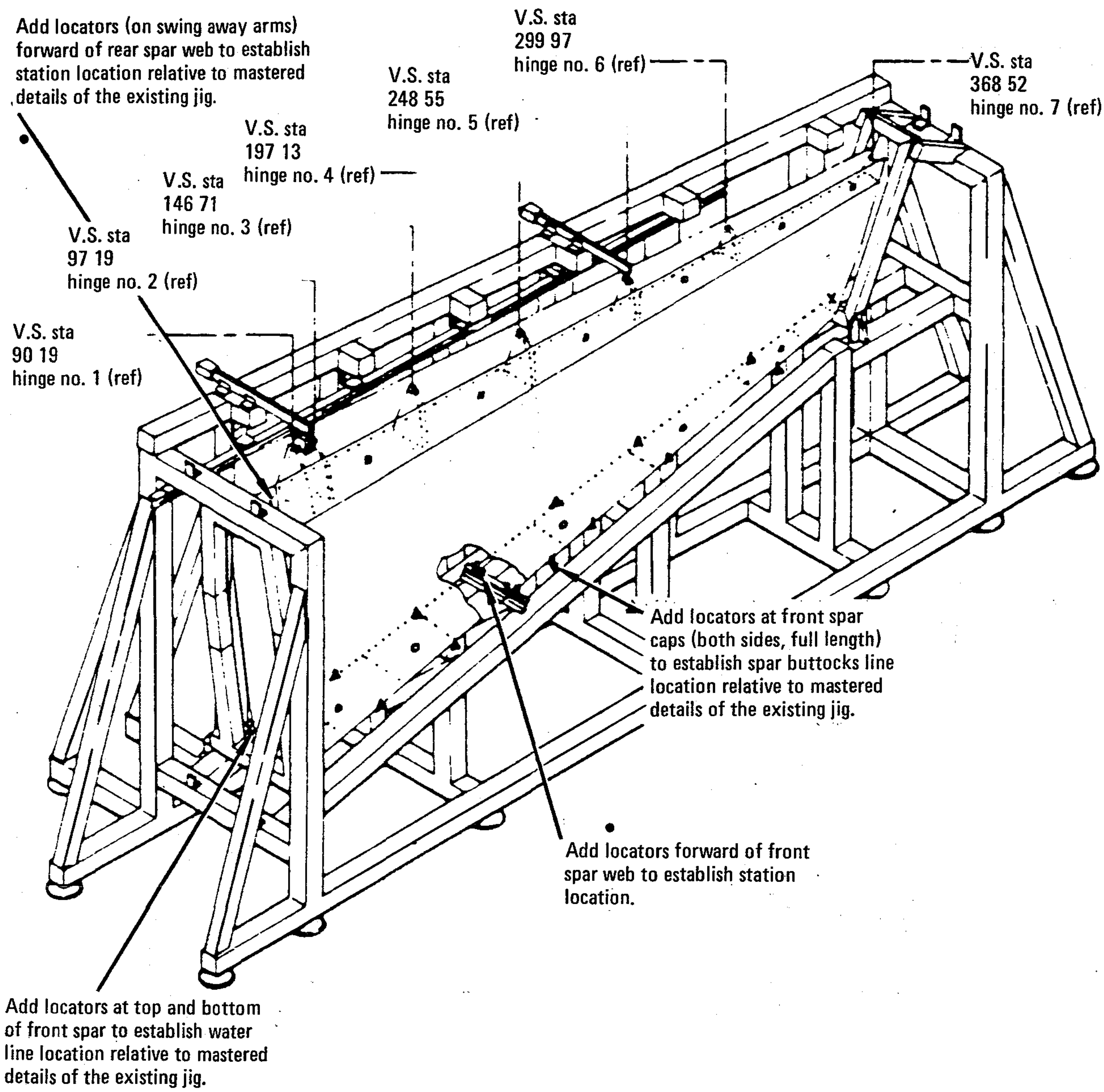


Figure 50. - Alterations to L-1011 fin assembly tool for ACVF.

5.2 Assembly Process

5.2.1 Subassembly of spars. - The assembly of the rear spar required 21 metal stiffeners, angles, clips, and fittings to be installed with approximately 500 fasteners. A similar number of parts and fasteners were used in the subassembly of the front spar. Metallic parts were painted with an epxoy primer and white, polyurethane top coat. Faying surface sealants were applied before permanently installing fasteners.

5.2.2 Assembly of fin box.- Assembly of the first L-1011 composite fin box began in January 1980. The covers and ribs were shipped from California in December of 1979 and arrived at Meridian, Mississippi, the first of January 1980. All components were inspected on receipt then prefitted and accepted for assembly.

Assembly of the first L-1011 composite fin box proceeded smoothly and efficiently. The first step in assembly of the fin box was to load the front spar in the assembly fixture as shown in figure 51.

Next, the ribs were loaded in the assembly tool as shown in figure 52 and temporarily attached to the front spar. Some minor discrepancies were found during fit-up, and these were corrected by minor rework of the rib flanges and by relocation of pilot holes.

After the ribs were loaded, the rear spar was installed as shown in figure 53.

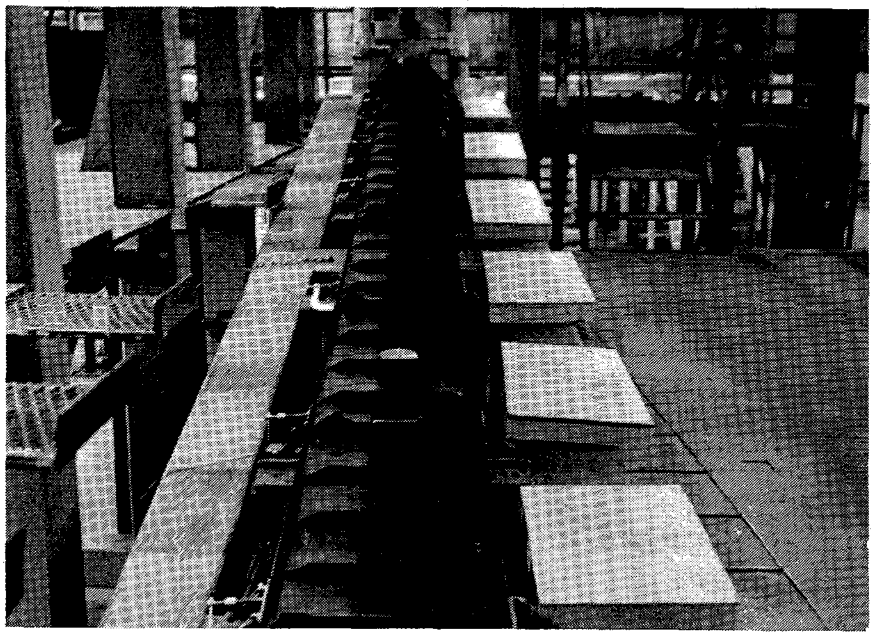


Figure 51.- Front spar loaded in fin box assembly fixture.

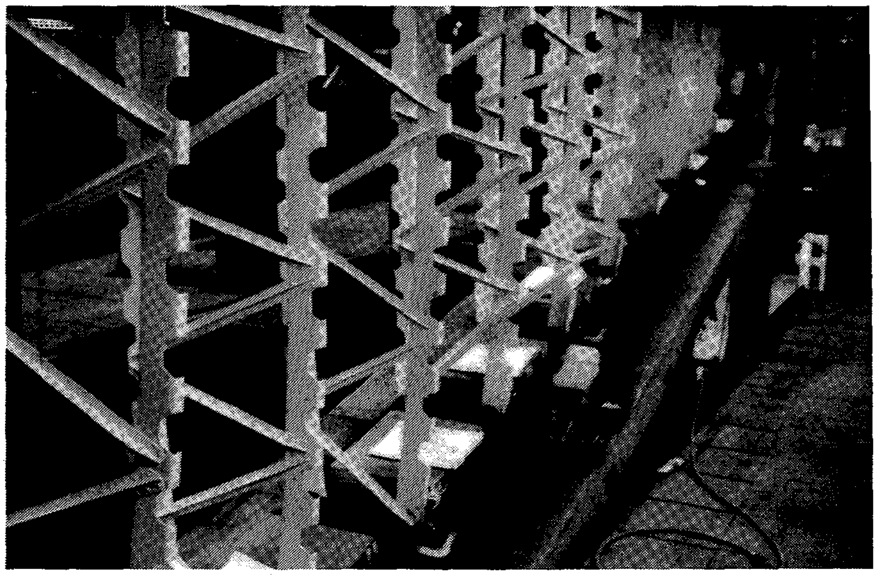


Figure 52. - Ribs loaded in assembly tool.

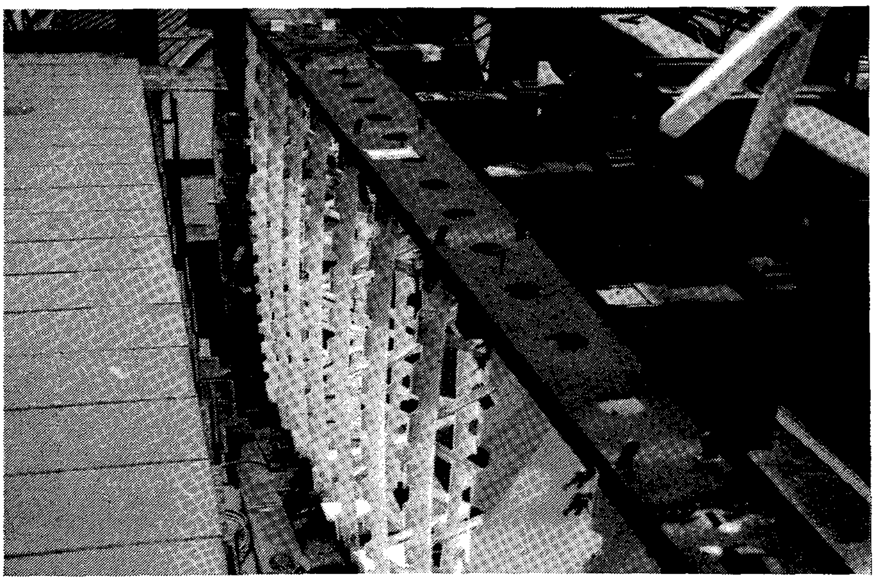


Figure 53. - Installation of rear spar.

The spars and ribs were permanently attached using titanium HL12 Hi-Loc pins and stainless steel HL94 collars. Kits containing metal parts and fasteners were supplied by Lockheed-California for each rib assembly, and these loose parts were used to complete the rib subassembly during assembly of the rib to the spar.

No significant problems were encountered during the assembly of the spars and ribs. Some very difficult areas in closed angles required angle drills to drill through the rib cap and rib attaching angle on the spar. All holes were inspected and accepted. The major problems that occurred during the assembly of the spars to the ribs were typical pilot hole mislocations, gaps, and design errors, and these discrepancies were not attributable to the composite structure.

Rudder actuator and hinge fittings were assembled to the ribs and rear spare as shown in figure 54. These fittings are used on the production L-1011 fins; they are in the same locations as those on the metallic fin and provide for interchangeability with the metallic rudder.

After the ribs and spars were attached permanently, the right-hand cover was loaded to the spar-rib framework, drilled and marked for trim. The temporarily installed right-hand cover was attached with clecoes as shown in figure 55. With the right-hand cover in place, the left-hand cover was

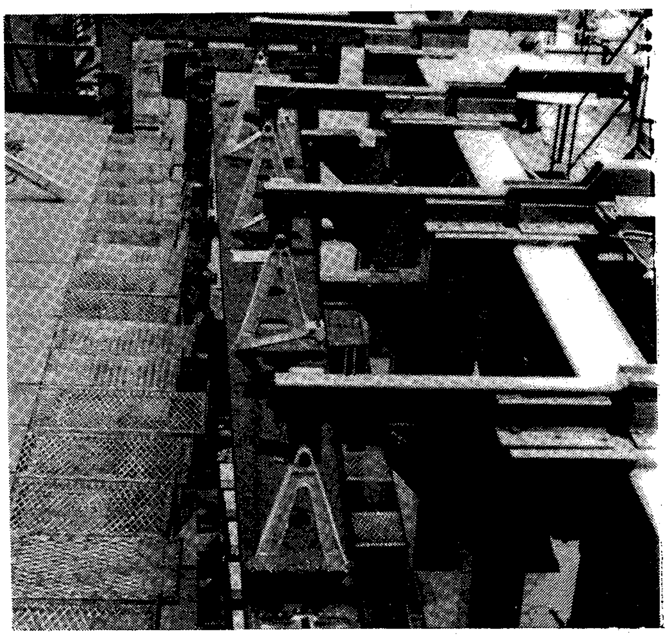


Figure 54. - Rudder actuator and hinge fittings.

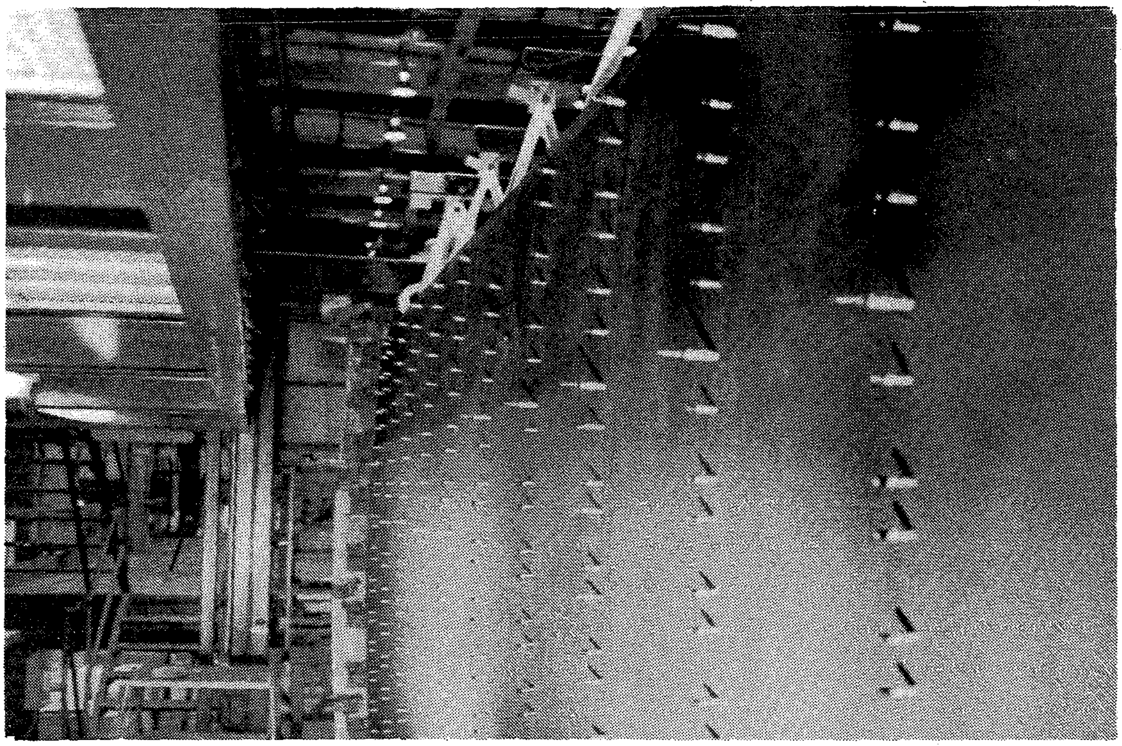


Figure 55. - Right-hand cover temporarily attached.

loosely loaded and marked. The right-hand cover was removed and trimmed while the left-hand cover was being drilled. The right-hand cover was reinstalled, and the left-hand cover was removed for trim. The right-hand cover was permanently attached with HL13V Hi-Loc titanium pins and HL94 stainless steel collars. The partially assembled fin box was turned over to instrumentation personnel for installation of strain gages before attaching the left-hand cover permanently.

Numerous small discrepancies typical of a first article fin assembly were uncovered, and practically all of these were in the metallic parts. No major problems were encountered. Discrepancies in the composite structure were easily disposed of with practically no loss of production time.

The left-hand cover was permanently installed with titanium pins, stainless steel collars, and MS21140 blind fasteners in the upper three solid ribs. Final close-out of the fin box required working inside the box as shown in figure 56. After completion of the assembly, the composite fin box was removed from the assembly fixture and placed on the support cradle. The completed fin box was loaded on a specially designed, shock-mounted dolly shown in figure 57 for shipment to Lockheed-California Company's test facility for ground testing.

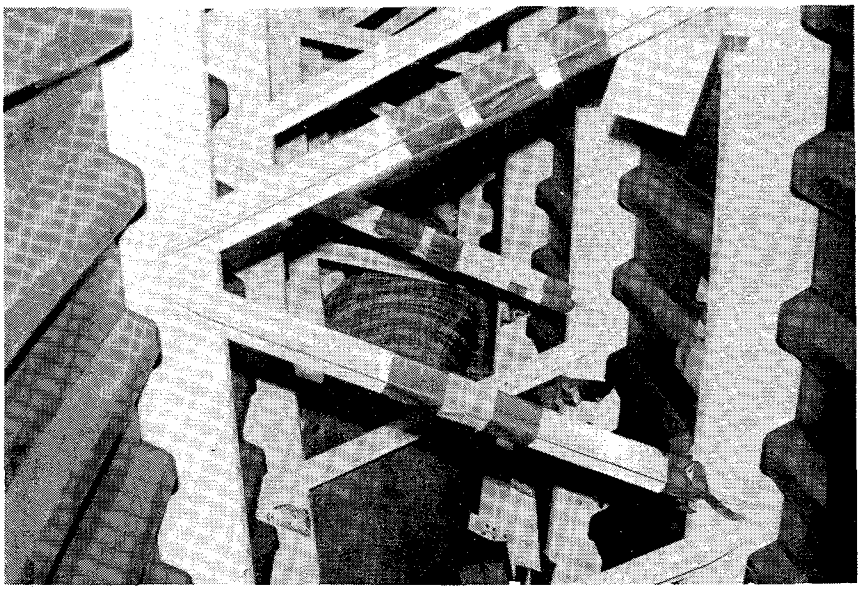


Figure 56. - Working inside fin box during final assembly.

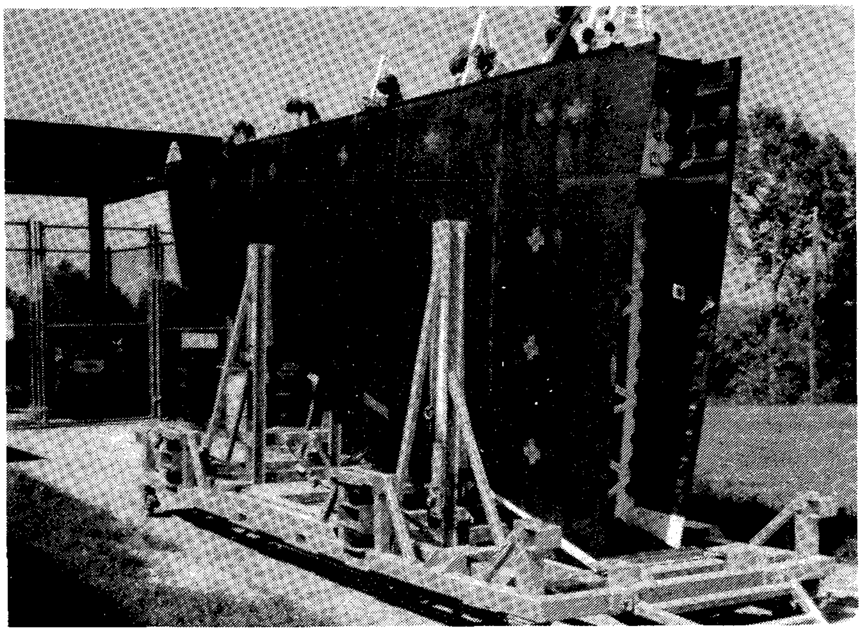


Figure 57. - Shipment of fin box on specially designed, shock-mounted dolly.

A second fin box was then assembled at Meridian. This box was identical to the first, both in assembly procedures and in instrumentation. Although start-ups, slow downs, and reschedules occurred during assembly of the second box, discrepancies and costs followed the typical drops expected on the learning curve of a second article.

5.3 Predicted and Actual Costs for Box Assembly

Early in the program, manufacturing estimated 3578 hours for the assembly of the first unit. This estimate was based on rigid requirements for drilling holes, hand fitting, and shimming gaps of 2.54 mm (0.01 inch) or greater. Following the manufacturing estimate, producibility/cost studies projected 1373 hours for assembly of the first production unit based on a production setup with hard tools, 100 units and a 78-percent learning curve. The first box actually required 1873 hours under a hand-made R&D assembly setup, and the second box required 1508 hours using the same setup. Figure 58 shows the original estimate made by Manufacturing for the R&D units, the actual hours for the first two L-1011 ACVF boxes, and the production estimate. The production estimate of 1373 man-hours is consistent with the actual times experienced.

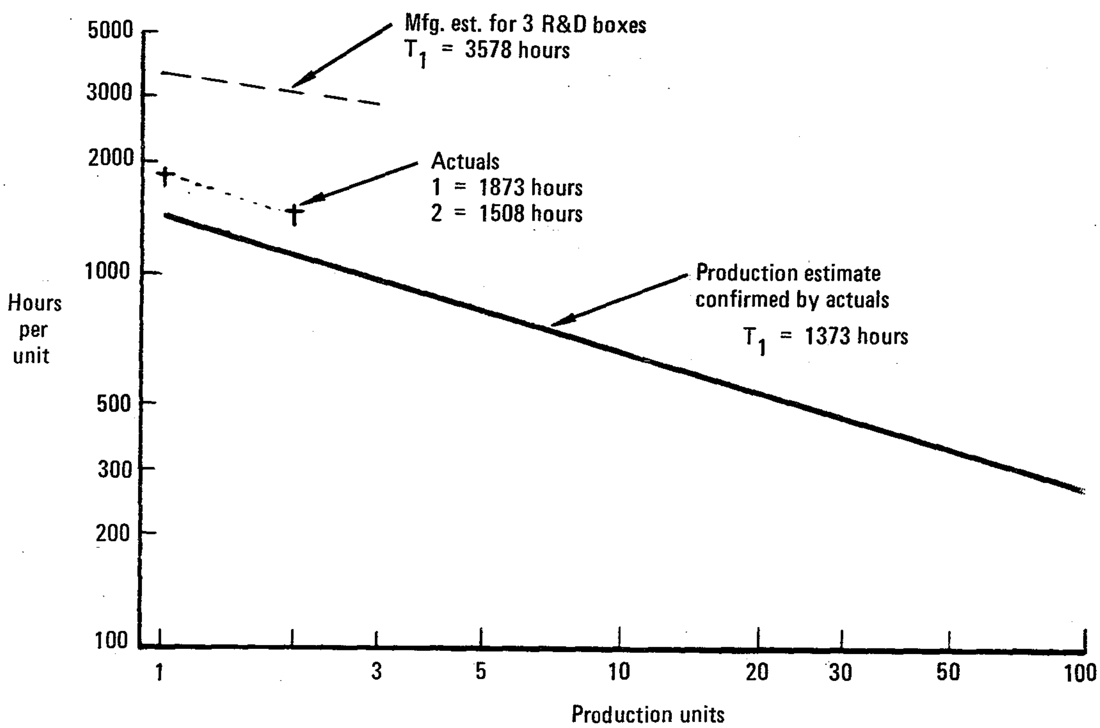


Figure 58. - Production box assembly estimates and actuals for first two L-1011 ACVF boxes.

6. QUALITY ASSURANCE

All quality assurance (QA) activity during Phase IV was carried out within the scope of the QA plan. The plan covered the activities of Lockheed-California Company as prime contract manager and Lockheed-Georgia Company as subcontractor, and elaborated on unique requirements to the ACVF. MIL-Q-9858A was used as the standard for QA at the two participating companies. The QA systems also conformed to federal airworthiness requirements. NAVPRO, the cognizant Government QA organization at Lockheed-California Company, had Government QA responsibility for the ACVF.

6.1 Traceability

The traceability requirements for all parts and assemblies were achieved through (1) supplier material certification, (2) material acceptance test certification, (3) in-process documentation on shop orders, and (4) FAA conformity certification. In-process traceability was achieved by recording material batch number and roll number on the applicable shop order. In addition, processing information of particular concern to ensure product integrity, such as material out time, and autoclave functions of temperature, time, and pressure, were recorded on the shop orders. All inspection tags processed by the Material Review Board were recorded on the shop orders and filed by the Inspection organization. The shop orders show a complete record of the production and inspection activities performed to produce the hardware and are retained as a permanent record of all detail parts and assemblies.

6.2 FAA Conformity Inspection

An FAA designated manufacturing inspection representative (DMIR) participated in the manufacturing operations by witnessing or monitoring all activities associated with the planning and fabrication or assembly of hardware, including the setups for testing. FAA Form 8130-9, Statement of Conformity, was completed by Manufacturing or Engineering for each component and assembly. After reviewing the form for accuracy, traceability, and completeness, the FAA DMIR prepared FAA Form 8100-1, Conformity Inspection Record, for each component and assembly. Both forms were subsequently forwarded to the FAA office.

6.3 Rejection History and Analysis

The following rejection history covers the fabrication of rib components and covers to the point where apparent changes in the graphite prepreg material caused process/cure problems. A tabulation of the various causes for rejection is shown in table 1. It should be noted that rejections caused by tooling and process development are included in the rejection statistics.

TABLE 1. - CAUSE FOR REJECTION

	VOIDS	FAILED LAB TEST	AUTOCLAVE FLAME OUT	VACUUM BAG BROKE	WORKMANSHIP	MISC.
TRUSS RIBS (2 CAPS)						
45 FABRICATED						
9 SCRAPPED		2	2	2	3	
SOLID WEB RIBS (1 PIECE)						
14 FABRICATED						
5 SCRAPPED	1	1	1		1	1
ACTUATOR RIBS (1 WEB, 2 CAPS)						
36 FABRICATED						
20 SCRAPPED	8	6		3	1	2
COVERS						
7 FABRICATED						
3 SCRAPPED	3					
TOTALS	12	9	3	5	5	3

6.3.1 Scrap rib components. - Workmanship rejections included such things as incorrect layup, incorrect trim, and mislocated holes. Corrective action in these instances was achieved by giving better instructions to manufacturing personnel and by providing foolproof tools. Process-related rejections included such things as broken vacuum bags, low resin content, low mechanical test results, and porosity and voids identified by nondestructive inspection (NDI). The problems of vacuum bag failures were virtually eliminated by removing sharp corners from the tooling and reinstructing manufacturing personnel. Adjustments were made during the process development phase to correct out-of-tolerance conditions noted by the test lab and NDI. Tooling errors were dimensional and affected either contour or trim. The tools were corrected, and subsequent parts were dimensionally correct. The final two actuator rib components were not refabricated because of material problems.

6.3.2 Scrap covers. - Ship 2 cover assembly was scrapped because of severe porosity and mark-off at the front and rear spar interface areas. The severe porosity problem was caused by a leak in the vacuum bag and a leak in an inflatable mandrel. The porosity areas were identified by ultrasonic inspection and confirmed by the QA laboratory after viewing core plugs with a 40x microscope. The problem was corrected by employing a double-bag procedure and performing high-pressure tests on the mandrels. The mark-off at front and rear spar interface area was caused by using several individual caul plates running along the front and rear spar interface areas. This condition was corrected by providing one-piece caul plates for both the front and rear spar areas.

Ship 3 cover-assemblies were both scrapped because of severe porosity. The porosity was due to an incoming material variation which was not detected during receiving inspection. As there was no test requirement for the third fin box, no replacement covers were fabricated.

7. COST ANALYSIS

One of the major goals of the ACVF program was to demonstrate cost competitiveness with the metal fin. Therefore, a comprehensive producibility/design-to-cost (DTC) plan was prepared outlining the basic steps necessary to achieve that goal. The aggressive execution of that plan resulted in a successful program and achievement of the goal. An important part of that task was to obtain manufacturing cost data in sufficient detail to establish the production costs and compare the data with prior projections.

Cost tracking for the DTC program was performed by the Engineering Branches of both Lockheed-California and Lockheed-Georgia. Manufacturing had the prime responsibility for the cost projections. Lockheed-California had responsibility for the covers, ribs, common items, and the overall DTC program. Lockheed-Georgia had responsibility for the spars and final assembly. The Phase III Production Readiness Verification Test Program PRVT provided useful information on cost trends and learning curves because of the more than 20 cover and spar components fabricated. Data obtained from Phase IV fabrication also contributed to the determination and verification of costs.

The results of the Producibility/Design-to-Cost program are summarized in table 2. Projected costs based on time standards are shown for the baseline metal fin, which is the target, and for the ACVF, using both existing manufacturing methods and automated methods. ACVF costs based on actuals using existing manufacturing methods are also shown. They represent the cumulative average cost for 100 units in 1983 dollars. The cost projections using existing manufacturing methods and the costs based on actuals have been refined and updated since the manufacturing review at Lockheed on 3 June 1980. ACVF existing manufacturing methods include hand layup, stacking, trimming, and

TABLE 2. – COST COMPARISON – SUMMARY

	CUM AVG COST FIRST 100 UNITS 1983 $	COST RATIO
• METAL FIN	$198,300	1.0
• ACVF – EXISTING MANUFACTURING METHODS	178,400	0.90
• ACVF – AUTOMATED MANUFACTURING METHODS	133,700	0.67
• ACVF – EXISTING MANUFACTUIRNG METHODS BASED ON ACTUALS	175,400	0.88

drilling. ACVF automated manufacturing methods involve numerical controlled tape layup, machine cutting, purchased preplied broadgoods, automated roll forming of hat stiffeners, platen press cured rib components, and powered gantry drilling for truss ribs.

The cost reductions compared to the metal fin using both existing and automated manufacturing techniques indicate the ACVF is cost competitive with metal. Automation of composites compared to existing manufacturing techniques indicate a cost reduction of 25 percent. The ACVF cost, based on actuals, is 12 percent under the metal fin. Facilities and equipment requirements for the automated manufacturing techniques are estimated to be approximately $12.1 million in 1983 dollars.

7.1 Final Cost Analysis

A final cost analysis was prepared incorporating all of the approved producibility cost reduction items* updating the cost estimates based on actuals, and refining the time standards, projections, and other cost criteria. The major component costs for the metal fin and the ACVF projections for existing and automated techniques are shown in table 3. Cost comparisons based on ACVF actuals to the metal fin and ACVF projected costs are also shown.

7.1.1 Cost comparison of ACVF using existing methods to design target. - A comparison of the ACVF (using existing manufacturing methods) to the metal fin indicates the following:

- The total cost of the ACVF is 10 percent below the design target.
- The cost of the covers is essentially equal.

TABLE 3. - COST COMPARISON BY COMPONENT (CUMULATIVE AVERAGE – 100 UNITS)

	(1) METAL FIN	(2) PROJECTED BASED ON EXISTING METHODS	(3) PROJECTED BASED ON AUTOMATED METHODS	(4) ACTUALS BASED ON EXISTING METHODS
COVERS	$ 66,000	$ 66,200	$ 42,500	$ 71,500
RIBS	40,200	28,800	19,400	24,500
SPARS	33,800	47,000	35,400	43,000
FINAL ASSEMBLY	53,300	31,400	31,400	31,400
COMMON ITEMS	5,000	5,000	5,000	5,000
	$198,300	$178,400	$133,700	$175,400
RATIO	1.0	0.90	0.67	0.88

*NASA CR-165634 Phase II Final Report Design and Analysis, April 1981.

- The cost of the ribs is approximately 28 percent less than the metal fin due primarily to design simplicity and reduction in the number of ribs, detail parts, and fasteners.

- The cost of the spars is approximately 39 percent greater than the metal fin. The weight reduction for the spars was approximately 44 percent.

- Final assembly costs for the ACVF are approximately 41 percent less than the metal fin.

- The cost for common items, including the auxiliary spar and the close-out rib that were retained in metal are equal for all cost projections.

7.1.2 (Cost comparison of metal fin to ACVF using automated methods to design target.) - A cost comparison of the ACVF using automated manufacturing methods to the metal fin, as shown in table 3, is summarized below:

- The total cost of the ACVF is approximately 33 percent less than the metal fin.

- The cost of the covers is 36 percent less than the metal fin.

- The rib costs are reduced by 52 percent.

- Spar costs are 5 percent greater than the metal fin.

- Final assembly costs are the same as those shown for existing methods reflecting a cost reduction of 41 percent.

- New facilities and equipment required for automation estimated at $12,109,000 in 1983 dollars are listed in table 4. The facilities and equipment would not be dedicated to the ACVF program only, but would be used for production of other components and programs.

7.1.3 Cost comparison of metal fin to ACVF based on actuals. - Using existing manufacturing methods, costs based on actuals for the ACVF are compared to the metal fin as shown in table 3. The costs for the ACVF based on actuals have been updated to include the latest information available. These costs are based on fabrication of the first three shipsets of components and assembly of the first two shipsets of components. First unit costs were established based on actuals and information obtained from the data abstraction forms as part of the cost tracking program.

First unit costs for the spars and final assembly are based on actuals for fabrication of the first three shipsets of spars and final assembly of the first two ACVFs. Actual cost data were used since none of the spars were scrapped and the actuals for the spars and final assembly were in line with

TABLE 4. - LABOR AND MATERIAL PROJECTED COST COMPARISON
METAL FIN VERSUS ACVF

MAJOR COMPONENT	METAL FIN: PROJECTED BASED ON TIME STANDARDS			ACVF: EXISTING METHODS			ACVF: AUTOMATED METHODS		
	LABOR	MATERIAL	TOTAL	LABOR	MATERIAL	TOTAL	LABOR	MATERIAL	TOTAL
COVERS	61,500	4,500	66,000	56,600	9,600	66,200	32,900	9,600	42,500
RIBS	37,700	2,500	40,200	25,200	3,600	28,800	15,800	3,600	19,400
SPARS	31,200	2,600	33,800	42,600	4,400	47,000	31,000	4,400	35,400
FINAL ASSEMBLY	50,300	3,000	53,300	27,700	3,700	31,400	27,700	3,700	31,400
COMMON ITEMS	4,800	200	5,000	4,800	200	5,000	4,800	200	5,000
TOTAL $	185,500	12,800	198,300	156,900	21,500	178,400	112,200	21,500	133,700
PERCENTAGE OF TOTAL DOLLARS	93.5	6.5	100.0	87.9	12.1	100.0	83.9	16.1	100.0
COST RATIO (1)	1.0	1.0	1.0	0.85	1.68	.90	.60	1.68	.67
COST RATIO (2)				1.0			0.72		

(1) Existing and automated methods for ACVF versus metal fin.
(2) Automated for ACVF versus existing methods for ACVF.

the cost projections. Costs included on the data abstraction forms for the PRVT and full-scale components were used in determining cost trends and learning curves. All costs were extrapolated to the fourth unit representing the first unit T_1 for a production program and appropriate learning curves were applied to arrive at the cumulative average for 100 aircraft.

First unit costs for the covers and ribs are based on costs taken from the data abstraction forms which represent hands-on or touch-time labor. A factor of 10 percent was added for personal time, fatigue, and delays. Appropriate learning curves were applied to the first unit costs to arrive at the cumulative average for the first 100 units. The actual factory time card hours for the covers and ribs were unusable for cost projection purposes for several reasons: the scrap rate was unusually high due to material problem unrelated to the manufacturing process. Time lags occurred to various manufacturing operations due to process development and refinement and engineering personnel forced to assist in some operations normally performed by production personnel in order to meet schedule. These situations are not unusual for a development program involving the use of an advanced material. Other circumstances that occurred during the manufacturing phase that further precluded the use of the time card actuals relating to a specified shipset (fin) serial number include: laying up, kitting, and storing of detail parts for more than one shipset at a time; and parts for more than one shipset were cured concurrently in the same autoclave.

Support costs for Quality Assurance, sustaining engineering, and sustaining tooling were expressed as percentages of production labor. Material costs included are the same as those projected costs using existing manufacturing methods.

The higher costs for the covers and spars are offset by lower costs for the ribs and final assembly, resulting in an overall cost reduction of 12 percent ($175,400 versus $198,300) compared to the metal fin.

- Cost for the covers is 8 percent higher based on actuals.
- Cost for the ribs is approximately 39 percent lower based on actuals.
- Cost for the spars is 27 percent higher based on actuals.
- Final assembly costs are 41 percent lower based on actuals.

7.1.4 <u>Cost comparison of ACVF based on actuals to projected costs</u>. - Using existing manufacturing methods, costs based on actuals are less than 1 percent higher than projected costs ($175,400 versus $178,400), as indicated in table 3. The derivation of the costs based on actuals is discussed in 7.1.3. The projected costs are primarily, based on time standards and manufacturing estimates.

Although the total costs are very close, there are some differences between components, which results from using two different methods of deriving costs. The differences between the cost based on actuals and the projected cost are discussed below. Since material costs are the same for both, the differences are in the labor costs.

- The cost of the covers is 8 percent higher ($71,500 versus $66,200) which is well within the accuracy required for estimating purposes.

- The cost of the ribs is 15 percent lower ($24,500 versus $28,800). The exact reasons for this difference are not known but the use of two different methods of estimating probably contribute to the cause.

- The cost for the spars is 8.5 percent lower ($43,000 versus $47,000).

- The cost for final assembly is equal for the same reasons given for the spars.

7.1.5 <u>Labor and material cost comparisons</u>. - Labor and material cost comparisons for each major component are shown in tables 4 and 5 for the metal fin and the ACVF. Toal component costs have been discussed previously.

Table 4 is a cost comparison of the metal fin and the ACVF using existing and automated manufacturing methods. The percentages of labor and material costs of the total dollars are shown, with cost ratios for the ACVF compared to the metal fin. [Total labor savings for the ACVF using existing and automated manufacturing methods are 18 and 40 percent, respectively.] The labor savings for the ACVF are achieved mainly through the reduced detail parts and fastener requirements of the ACVF configuration and design characteristics. Automated manufacturing methods for the ACVF reduce labor cost approximately 27 percent over the use of existing manufacturing methods.

Material costs are 68 percent higher for the ACVF than the metal fin because of the higher composite raw material costs ($20 per pound); higher scrap factors (15 percent for the covers and ribs, 30 percent for the spars, versus 9.5 percent for the metal fin); and the use of titanium HiLok fasteners versus aluminum rivets for the metal fin. Although material costs for each major component of the ACVF are higher than the metal fin (more than double for the covers), the reduced labor costs more than offset the material cost increase with the exception of the spars. The spars for the ACVF are always higher than the metal spars due to the ACVF configuration and manufacturing processes involved. The weight reduction of approximately 44 percent more than justifies the use of composites for the spars on a cost per pound of weight saved basis.

Table 5 is a comparison of projected costs and costs based on actuals for the ACVF using existing manufacturing methods. Labor and material costs are shown for each major component or task. The labor and material percentages are identical as they contribute to the total dollars, even though the total dollars are slightly different. The labor costs based on actuals are approximately 9 percent higher for the covers ($61,900 versus 56,600) and 17 percent

TABLE 5. - LABOR AND MATERIAL PROJECTED COST COMPARISON
PROJECTED ~ ACTUALS

MAJOR COMPONENT/TASK	ACVF EXISTING METHODS BASED ON PROJECTED COSTS			ACVF EXISTING METHODS BASED ON ACTUALS		
	LABOR	MATERIAL	TOTAL	LABOR	MATERIAL	TOTAL
COVERS	$ 56,600	$ 9,600	$ 66,200	$ 61,900	$ 9,600	$ 71,500
RIBS	25,200	3,600	28,800	20,900	3,600	24,500
SPARS	42,600	4,400	47,000	38,000	4,400	43,000
FINAL ASSEMBLY	27,700	3,700	31,400	27,700	3,700	31,400
COMMON ITEMS	4,800	200	5,000	4,800	200	5,000
TOTAL	$156,900	$21,500	$178,400	$153,900	$21,500	$175,400
PERCENTAGE OF TOTAL DOLLARS	87.9	12.1	100.0	87.7	12.3	100.0
COST RATIO	1.0	1.0	1.0	0.98	1.0	0.98

lower for the ribs ($20,900 versus $25,200) than the projected costs. This difference is not surprising considering two completely different methods of estimating were used. The cost for the spars is 11 percent lower ($38,000 versus $42,600).

7.2 Sensitivity Studies

Several sensitivity studies were conducted to determine the cost impact of varying certain criteria. The results of these studies are summarized in table 6. The QA hours were calculated using 30 percent instead of 15 percent of production labor hours resulting in a cost increase of $17,000. The material scrap factor for the covers and ribs was increased from 15 to 30 percent with a cost impact of $1,400. The basic graphite-epoxy material cost was calculated using $30 instead of $20 per pound. This would add $7,000 to the ACVF.

The cumulative effect of these increases, totaling $25,400, would result in a total of $203,800 for the ACVF using existing methods. This is slightly less than 3 percent over the metal fin ($198,300), which is still competitive considering the weight saved. The total increase of $25,400 added to the ACVF using automated techniques of $133,700 results in a total cost of $159,000, or approximately 20 percent below the metal fin.

7.3 Production Readiness Verification Test Cost Data

The cost data for the PRVT cover and spar articles were used in performing extrapolations to arrive at the projected costs for the full-size components. No PRVT articles were made for the ribs.

7.3.1 Production readiness verification test. - Each PRVT cover consists of the following components:

- Skin
- Stiffeners

TABLE 6. - SENSITIVITY STUDIES

ITEM	BASE VALUE	ALTERNATE VALUE	IMPACT
QUALITY ASSURANCE	15%	30%	$17,000
MATERIAL SCRAP	15% LOCKHEED-CALIFORNIA CO. 30% LOCKHEED-GEORGIA CO.	30%	1,400
MATERIAL COST	$20/LB	$30/LB	7,000

- Fillers
- Doublers

These components are assembled together and cocured into an integral unit. Assigned observers have recorded the number of man-hours expended for both fabricating and assembling each of the above components at a manufacturing process/operation level. Because of problems encountered during assembly and cocuring, 27 units were produced, 20 of which were acceptable. The total time for assembling and curing the rejected covers was not reported and the data are no longer available. In order to derive a learning curve for the actual time required to produce 20 covers, the time expended on the rejected covers must be included. Since this time is not available, an alternate approach has been applied.

The fabrication time for the stiffeners, fillers, and doublers on each of the 27 units was tracked during the manufacturing process/operation level (i.e., tool preparation, composite orientation, pattern cutting, layup, etc.), as required for the Structural Composite Fabrication Guide/Data Abstraction form. Based on these data, which represent approximately 50 percent of the total man-hours for each cover, a credible learning slope has been determined. On the basis of learning curve theory, each time the total quantity of items produced doubles, the cost per item is reduced to some constant percentage of the previous costs.

The following tracked man-hours for five sets of cover details were used to determine the learning curves.

PRVT Cover details set No. 2 equals 42.1 man-hours

PRVT Cover details set No. 4 equals 40.1 man-hours

PRVT Cover details set No. 8 equals 38.1 man-hours

PRVT Cover details set No. 16 equals 32.4 man-hours

PRVT Cover details set No. 27 equals 28.0 man-hours

As shown in figure 59, sets (2, 4 and 8) indicate the 95-percent slope. Using sets (8, 16 and 17) an 82-percent slope is indicated.

A computer analysis of the five sets in relation to the learning slope line indicates an overall 84 percent slope, which confirms the learning curve that was previously selected based on other rationale and which is stated in the premises.

The actual man-hours required to fabricate the 20 PRVT spars were tracked both by unit and operation. Table 7 gives the total time card hours by

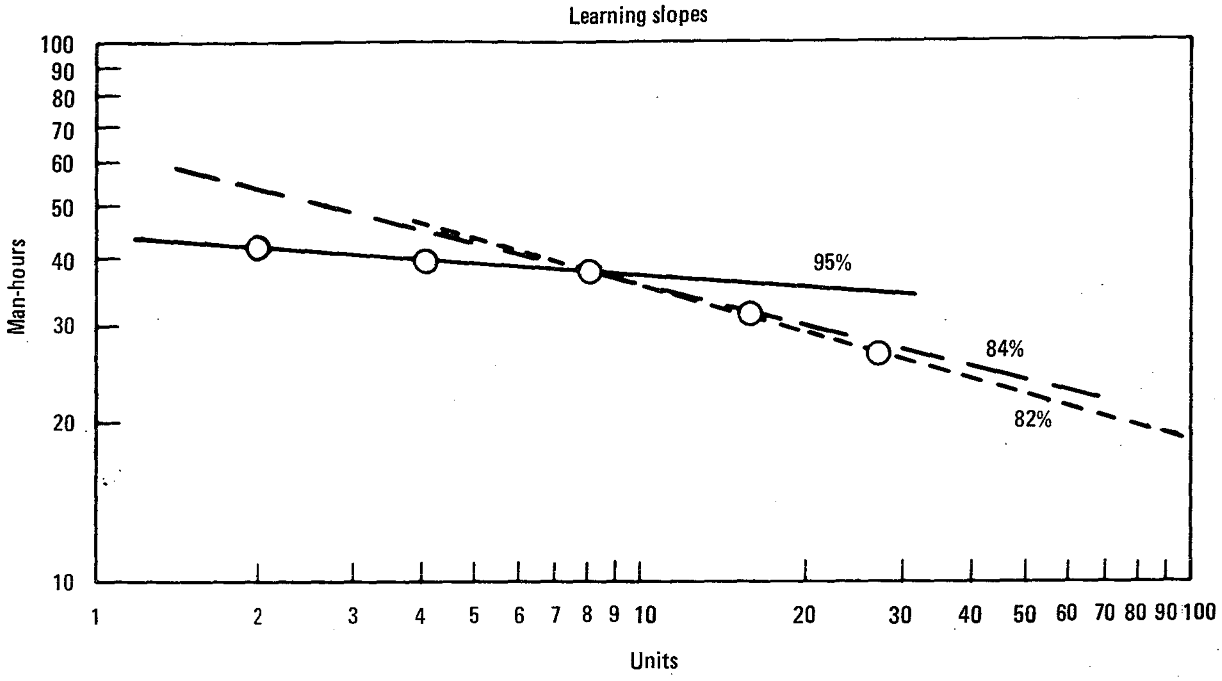


Figure 59.- L-1011 ACVF/PRVT covers.

operation for the Structural Composite Fabrication Guide/Data Abstraction Form. Analysis of the data gives insight in to the effects of the learning achieved for the 20 units by operation and enables an estimator to more accurately predict the overall learning curve slope that will be achieved in production.

The total fabrication hours were plotted as shown in figure 60. An 83-percent learning curve was achieved for the 20 PRVT spars. Eliminating spars 11 and 12 from the data, changes the slope to 82 percent and agrees with the original projection. The PRVT data was further extended to compare with the production spar fabrication estimates and is also shown in figure 60. The adjusted full-length curve falls slightly below the projected curve. This result was considered close enough not to warrant any changes to the original production estimates used in the Design-to-Cost Report. The adjusted full-length curve as noted in figure 60 was determined simply by the ratio of weight for the PRVT spars and full length spars and adjusting for automation and the production environment.

7.4 Cost Tracking – Data Accumulation and Recording

Data for inclusion into the Structural Fabrication Guide for Advanced Composites, AFML/LTN (Contract No. F33615-75-C-5009), was tracked and documented. Each identifiable composite part and/or assembly and PRVT specimen

TABLE 7. - PRVT SPAR FABRICATION HOURS.

PRVT NO.	TOOL PREPARATION	COMPOSITE ORIENTATION	PATTERN CUTTING	ASSEMBLE DETAILS	INSTRUMENT	BAGGING AND TOOL CLOSURE	CURE	POST-CURE	PART REMOVAL	DRILL TRIM	TOTAL
1	65	40	80	62	13	85	18	10	20	15	408
2	65	38	42	39	12	80	18	10	18	9	331
3	62	22	36	35	12	78	18	10	16	9	298
4	59	19	31	30	10	76	18	10	16	8	277
5	56	11	22	20	10	76	18	10	16	8	247
6	56	19	39	20	10	35	18	10	14	9	230
7	79	17	35	20	9	15	19	10	12	9	225
8	76	16	33	19	6	14	19	10	10	9	212
9	80	15	29	20	5	19	19	10	10	10	217
10	81	15	29	21	5	20	19	10	10	10	220
11	88	22	40	24	7	43	19	10	9	10	272
12	75	18	38	25	8	38	19	10	9	10	250
13	77	11	33	26	4	14	19	11	8	11	214
14	74	11	37	24	3	11	19	12	10	9	210
15	58	12	35	46	4	10	19	12	8	10	214
16	51	12	30	28	4	8	19	15	4	12	183
17	48	12	30	27	3	8	19	7	4	6	164
18	45	12	30	27	3	7	19	6	3	8	160
19	42	12	30	26	3	8	20	5	4	4	154
20	47	12	30	27	3	8	20	5	4	4	160

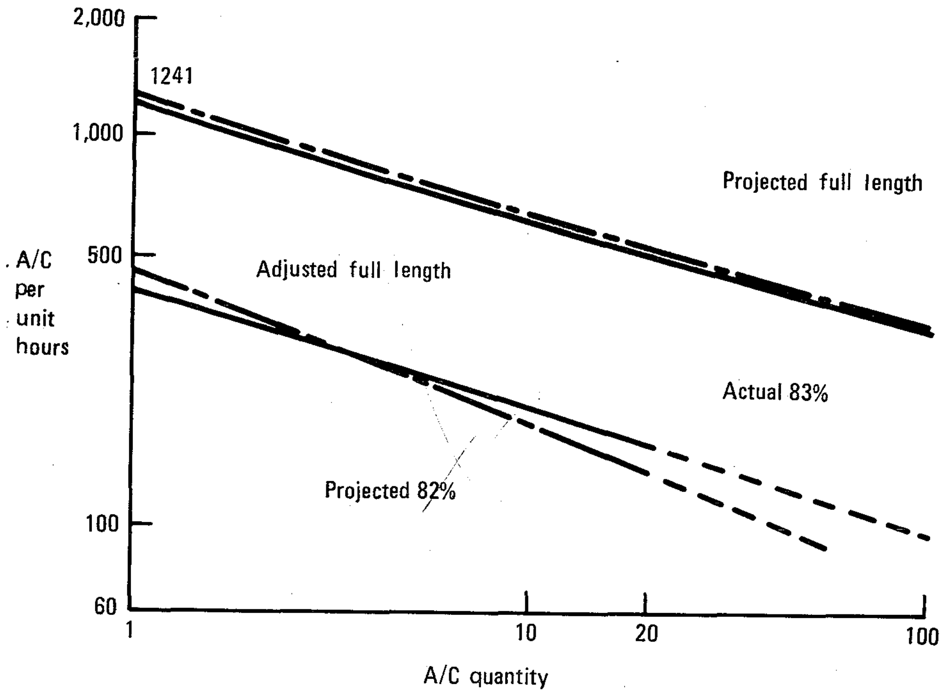


Figure 60. - Composite spar fabrication – actual versus estimated.

was recorded and reported on the data abstraction forms. The tracking, in addition to stating the purpose and complete identification of the component, included structural application, material, primary tooling, part construction, cost, fabrication techniques, quality control, and experience information.

A composite component in this context is any composite part that has been produced in any quantity, identified with a part number, cured, and can be stocked in a nonrefrigerated area until its evential use. The information on the Data Abstraction form was supplied by the appropriate division, i.e., Manufacturing, Quality Control, etc., then formalized and submitted by the Lockheed-California Company to NASA.

The composite component cost and material usage data for the Data Abstraction forms were derived from man-hours and weights manually recorded on the physical shop order at the conclusion of the manufacturing process by the operator in the shop. These data were recorded on the form shown in figure 61, CME Bulletin No. 274, which is a sample showing the format.

After review and corrections as required by supervision, the cost/weight information was forwarded to Engineering for transcription to the data abstraction form and eventual transmittal to NASA.

LOCKHEED

C M E – MANUAL

LOCKHEED – CALIFORNIA COMPANY
A Division of Lockheed Aircraft Corporation

LOCKHEED

1608289-125
UNIT #20

CME BULLETIN #274
REVISION #1

TO: All Planning Personnel

SUBJECT: COMPOSITE VERTICAL FIN PROGRAM

GENERAL

Existing CME Procedures will be complied with unless specifically authorized for deviation by this Bulletin. This Bulletin will be updated and reissued as deviation requirements occur.

A principal objective of this program is to acquire data on the costs of manufacturing composite structure components.

In addition, the processes and costs will be carefully documented and used to estimate the costs of similar components manufactured in a production environment.

* Complete cost records shall be maintained for all Phase III fabrication effort. Operation Sheets, Tool Orders, and Production Inspection Records (PIR's) shall include the following labor, material, and weight recording provisions as applicable.

A. Direct Fabrication Labor (Manhours/Unit)

Major Operation		Manufacture	Quality Assurance
1.	Tool Preparation	.3	.1
2.	Composite Orientation	1.3	.7
3.	Pattern Cutting	.4	.2
4.	Layup Detail Installation	.5	.3
5.	Honeycomb Preparation		
6.	Instrumentation*		
7.	Bagging/Tool Closure		
8.	Cure/Consolidation		
9.	Post Cure		
10.	Part Removal Cleanup	.3	.1
11.	Machining/Trimming/Drilling	.4	.2
12.	Finishing	.4	.2
13.	Assembly	.6	.3

B. Dimensions:

1.	Maximum Width	2	Inches	50.8 MM
2.	Maximum Thickness	.010	Inches	.250 MM
3.	Maximum Length	36	Inches	914.4 MM
4.	Wetted Area		Inches	
5.	Outside Diameter		Inches	
6.	Wall Thickness		Inches	
7.	Taper Ratio		Inches/Inch	

ISSUED: 10-24-75
Rev. Eff.: 1-27-76

* Revised
** Added
*** Deleted

CME BULLETIN #274
REVISION #1

Page 1 Of 2

FORM 6848-6

Figure 61. - CME bulletin No. 274.

1608289-125
UNIT #20

C. Weight:

1. Raw Material Purchased for Part ________Pounds
2. Completed Part ________Pounds (include non-composite)
3. Non-composite Part ________Pounds
4. Trimmed Material ________Pounds
5. Test Material ________Pounds
6. Consumable Materials ________Pounds
7. Assembly ________Pounds

D. Ply Information:

1. Number of Plies 2 Maximum 2 Minimum
2. Ply Orientation + –

E.

1. Time Removed from Freezer 7 AM Date 3-27-79
2. Total Elapsed Time at Room Temperature 30 HRS.

Planning is responsible for reflecting all configuration control requirements, as determined by Engineering, into work authorizing documents on a planned basis.

Tool Orders for tooling controlling critical dimensions and/or interfaces will carry the statement, "Tool verification mandatory." The interface control drawing will be referenced.

Tool Orders, Production Inspection Record Books, and Manufacturing Change Notices will be overstamped, "Componsite Fin."

Figure 61. - Concluded.

7.5 Cost Summary

The results of this program in terms of cost indicate that composite structures are competitive with metal. The cost projections resulting from this program indicate savings of 12 and 33 percent when comparing existing and automated manufacturing methods for the ACVF to the metal fin. The reduced number of parts and fasteners is the most likely reason for reduced costs. The ACVF cost based on actuals is less than 1 percent higher than the projected cost using existing manufacturing methods.

Establishing a comprehensive Design-to-Cost plan, complying with the detailed requirements of the plan, establishing a producibility team, and rigorously conducting a producibility/cost reduction program resulted in meeting the program goal of making composites cost competitive with metal. Although the spars are not cost competitive with metal, even with the use of automated manufacturing methods ($35,400 versus $33,800), the cost increase of less than 5 percent was acceptable based on a weight savings of 44 percent.

Several sensitivity studies were conducted involving QA requirements, basic graphite-epoxy material costs, and scrap factors. These cost excursions provide the cost impact of changes to the premises and assumptions. The cumulative effect of the sensitivitity studies, added to the ACVF using automated manufacturing methods, produced results that were approximately 20 percent below the metal fin. Parametric cost data in terms of hours and material dollars per pound and cost ratios can be developed for use on other similar structures and/or programs.

8. CONCLUSIONS

Two complete fin boxes were fabricated and assembled demonstrating that advanced composite primary structures are cost competitive with metal structures. Fabrication and assembly was accomplished in a production environment and experienced no special problems other than those normally encountered with a new production program.

Valuable lessons were learned in tool design and processing.

BIBLIOGRAPHY

Eudaily, R.R., "Assembly of the L-1011 Composite Fin," Presented at the Fifth Conference on Fibrous Composites in Structural Design, New Orleans, Lockheed-California Company, January 1981.

Rogers, A.K., Tajima, Y.A., Young, R.C., "Material Characterization and Specification Development for 350°F Curing Epoxy-Graphite Materials, to be presented at the 27th National SAMPE Symposium, San Diego, Lockheed-California Company, May 1982.

Tajima, Y., Rogers, A., "The Development of a Specification for 350°F Curing Epoxy-Graphite Materials," presented at the ASTM Conference for Producibility and Quality Assurance of Composite Materials, St. Louis, Lockheed-California Company, October 1981.

1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.

4. Title and Subtitle

ADVANCED MANUFACTURING DEVELOPMENT OF A COMPOSITE
EMPENNAGE COMPONENT FOR L-1011 AIRCRAFT

5. Report Date
5-4-82

6. Performing Organization Code

7. Author(s)
T. Alva, J. Henkel, R. Johnson, B. Carll, A. Jackson,
B. Mosesian, G. Brozovic, R. O'Brien, R. Eudaily, et.al.

8. Performing Organization Report No.

9. Performing Organization Name and Address

10. Work Unit No.

11. Contract or Grant No.
NAS1-14000

12. Sponsoring Agency Name and Address

13. Type of Report and Period Covered
Contract Report

14. Sponsoring Agency Code

15. Supplementary Notes

16. Abstract

This is the final report of technical work conducted during the fourth phase of a multiphase program having the objective of the design, development and flight evaluation of an advanced composite empennage component manufactured in a production environment at a cost competitive with those of its metal counterpart, and at a weight savings of at least 20 percent. The empennage component selected for this program is the vertical fin box of the L-1011 aircraft. The box structure extends from the fuselage production joint to the tip rib and includes front and rear spars.

During Phase IV of the program, production quality tooling was designed and manufactured to produce three sets of covers, ribs, spars, miscellaneous parts, and subassemblies to assemble three complete ACVF units. Recurring and nonrecurring cost data were compiled and documented in the updated producibility/design-to-cost plan. Nondestruct inspections, quality control tests, and quality acceptance tests were performed in accordance with the quality assurance plan and the structural integrity control plan. Records were maintained to provide traceability of material and parts throughout the manufacturing development phase. It was also determined that additional tooling would not be required to support the current and projected L-1011 production rate.

17. Key Words (Suggested by Author(s))

18. Distribution Statement
~~FEDD~~ and Subject
Category 24

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
UNCLASSIFIED	UNCLASSIFIED	75	

N-305

Available: NASA's Industrial Applications Centers

End of Document