DESIGN, ANALYSIS AND FABRICATION OF THE TECHNOLOGY INTEGRATION BOX BEAM

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SUMMARY

Numerous design concepts, materials, and manufacturing methods were investigated analytically and empirically for the covers and spars of a transport wing box. This information was applied to the design, analysis, and fabrication of a full-scale section of a transport wing box.

A blade-stiffened design was selected for the upper and lower covers of the box. These covers have been constructed using three styles of AS4/974 prepreg fabrics. The front and rear T-stiffened channel spars were filament wound using AS4/1806 towpreg. Covers, ribs, and spars were assembled using mechanical fasteners. When they are completed later this year, the tests on the technology integration box beam will demonstrate the structural integrity of an advanced composite wing design which is 25 percent lighter than the metal baseline.

INTRODUCTION

Current applications of composite materials to transport aircraft structure, most of which are stiffness critical secondary structural components, have demonstrated weight saving from 20 to 30 percent. The greatest impact on aircraft performance and cost will be made when these materials are used for fabrication of primary wing and fuselage structures that are 30 to 40 percent lighter than their metal counterparts and have a reduced acquisition cost. Achievement of this goal requires the integration of innovative design concepts, improved composite materials, and low cost manufacturing methods.

In 1984, the Lockheed Aeronautical Systems Company began a program to develop engineering and manufacturing technology for advanced composite wing structures on large transport aircraft. The program was sponsored by the National Aeronautics and Space Administration (NASA) under contracts NAS1-17699 and NAS1-18888 and Lockheed Aeronautical Systems Company independent research and development funds.

The selected baseline component is the center wing structural box of an advanced version of the C-130 aircraft. The existing structural box, shown in Figure 1, is a two-spar multi-rib design, 440 inches long, 80 inches wide, and 35 inches deep at the crown. A preliminary design of a composite wing box was completed as were many design development tests. A full-scale section of the wing box was designed in detail, analyzed, and fabricated. This paper will summarize the major technical achievements of the box beam program.

It should be noted that some concessions as listed herein were made to reduce the cost of the program; the conclusions drawn thus far in this program are valid and achievable.

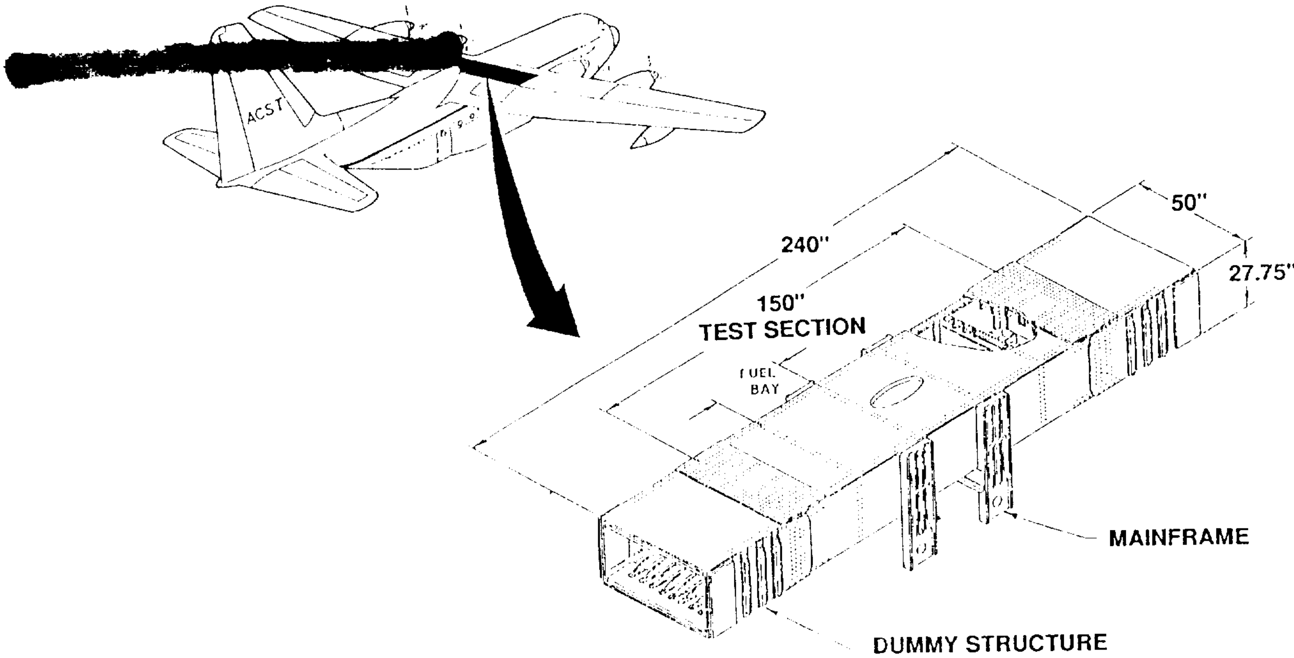


Figure 1. Technology Integration Box Beam

- Surface contour was eliminated
- Same tooling used for upper and lower covers
- L/E and T/E attachments were omitted
- Ribs were made of aluminum
- Cut-outs in the spars were omitted
- Stiffeners on spars were made of aluminum
- Constant sections were used
- Hand lay-up of hat-stiffeners, doublers, and C-channels was used
- Additional autoclave cycles were used for doublers
- Both spars were filament wound at once on common mandrel
- Aluminum access door hole doubler was bolted
- Spar/cover joint doublers were secondary bonded

BOX BEAM DESIGN AND ANALYSIS

GEOMETRY

The technology integration box beam, see Figure 1, represents a highly loaded full-scale section of the C-130 center wing box. The test section of the box is 150 inches long, 50 inches wide, and 28 inches deep. The test section contains a large access hole in the upper cover, wing box-to-fuselage mainframe joints, and center wing-to-outer wing joints.

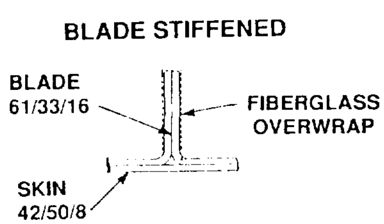
DESIGN LOADS AND CRITERIA

The design loads for the box beam are based on the baseline aircraft requirements. Maximum ultimate loads are 26,000 lb/inch compression in the upper covers and 24,000 lb/inch tension in the lower covers. Ultimate spar web shear flow is 4,500 lb/inch. These loads are combined with the appropriate pressure loads due to beam bending curvature and fuel. The stiffness requirements for the wing were established to meet the commercial flutter requirements specified in FAR Part 25. Stated briefly, at any wing station the composite wing bending stiffness and torsional stiffness could not be less than 50 percent of the baseline wing, and the ratio of the bending to torsional stiffness must be greater than one but not more than four.

Structural requirements for damage tolerance considered civil as well as military criteria. Thus, the criteria used for this program requires the structure to have ultimate strength capability with the presence of barely visible impact damage anywhere within the structure. Barely visible impact damage is either the kinetic energy required to cause a 0.1 inch deep dent or a kinetic energy of 100 ft-lb with a 1.0 inch diameter hemispherical impactor, whichever is least.

COVER DESIGN

Design trade studies and structural tests were conducted on various configurations for the wing covers. Figure 2 describes several of the designs and presents results of compression tests on impact damaged panels. Based on these investigations and manufacturing cost estimates, the blade stiffened design was selected for the box beam.



DESIGN	MATERIAL	COMPRESSION FAILURE LOAD / AREAL WEIGHT (IN.) ①
BLADE	IM7/8551-7E (TAPE)	940,000
BLADE	AS4/2220 (TAPE)	590,000
T	AS4/1806 (TAPE)	810,000
J	AS4/ 1806 (FABRIC)	800,000

① 100 FT-LB IMPACT

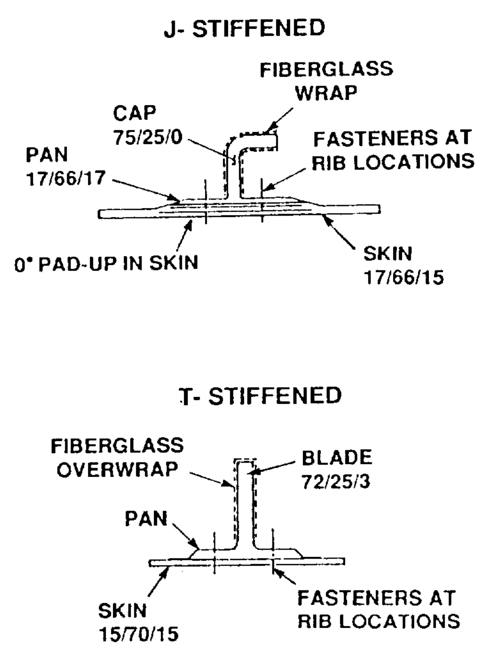


Figure 2. Wing Cover Designs

The lower cover design, shown in Figure 3, consists of back-to-back channels laid up on a skin laminate to form a blade stiffened panel. Note that the flanges of the channels contain additional 0 degree plies compared to the web thus resulting in a blade containing 67 percent 0 degree plies, 29 percent plus/minus 45 degree plies, and 4 percent 90 degree plies. The blades, which are spaced at 5 inches, are tapered in height to account for the increased axial loading from the outboard joint to the wing centerline. A constant thickness laminate containing 27 percent 0 degree plies, 64 percent plus/minus 45 degree plies, and 9 percent 90 degree plies makes up the lower skin.

The configuration of the upper cover, shown in Figure 4, is similar to the lower cover with the exception that the blades are slightly taller. Also, the central bay of the upper cover is reinforced by a hat stiffener which is terminated at each rib location. The cut-out in the upper cover is reinforced by an aluminum plate which is mechanically fastened to the cover laminate.

The chordwise splice of the composite covers to the aluminum load introduction box is accomplished with the double shear joint illustrated in Figure 5. Note, the bending stiffness continuity of the cover is maintained by inserting the aluminum splice Ts between the composite blades.

SPAR DESIGN

As with the covers, trade studies and subcomponent tests were conducted on various spar designs. Figure 6 shows the results of tests on stiffened shear panels manufactured using several different materials and methods. The results of these studies when combined with manufacturing cost estimates led to the selection of the T-stiffened channel configuration shown in Figure 7. The spar webs and caps are of constant thickness with the exception of the doublers located at the mainframe attachment and the spar splice locations. This spar was designed to be filament wound using AS4/1806 towpreg with unidirectional, bidirectional, and bias fabrics used for the spar cap inserts, and doublers. The stiffeners were made of aluminum for economy and were bolted and bonded to the spar web.

RIBS AND BOX ASSEMBLY

For the box beam, a J-stiffened skin configuration constructed of aluminum was selected for all of the ribs. As shown in Figure 8, a T-shaped shear tie connects the rib web and rib cap to the cover. All ribs will be mechanically fastened to the spar webs and covers. Also the spar caps are mechanically fastened to the covers using a double row of fasteners.

STRUCTURAL ANALYSIS

A detailed structural analysis was completed on the box beam using the methods shown in Figure 9. A three-dimensional finite element model was constructed and used to obtain internal loads for sixteen loads cases. Detailed two-dimensional models were used to analyze the cover chordwise joint, cover cut-out area, and the mainframe to spar joint. The compression stability of the covers was predicted using the PASCO code obtained from NASA. Several Lockheed computer programs were used to obtain local stresses and strains using the internal loads obtained from the NASTRAN models.

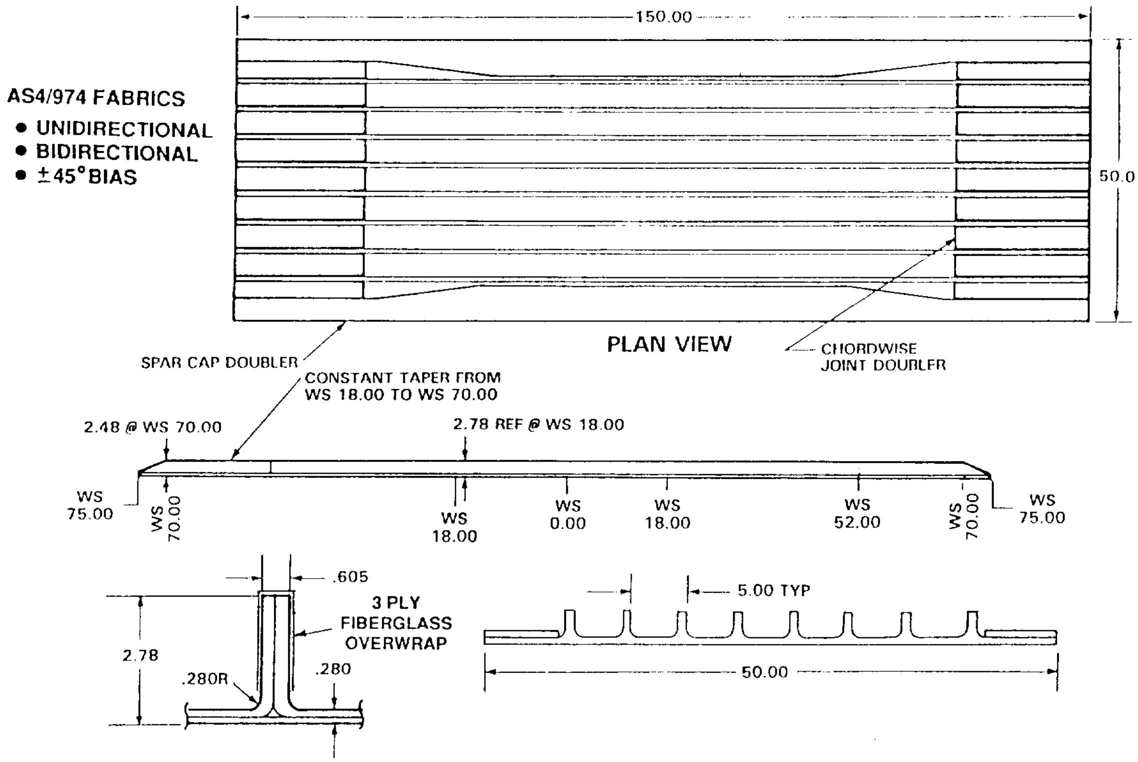


Figure 3. Lower Cover Box Beam Design

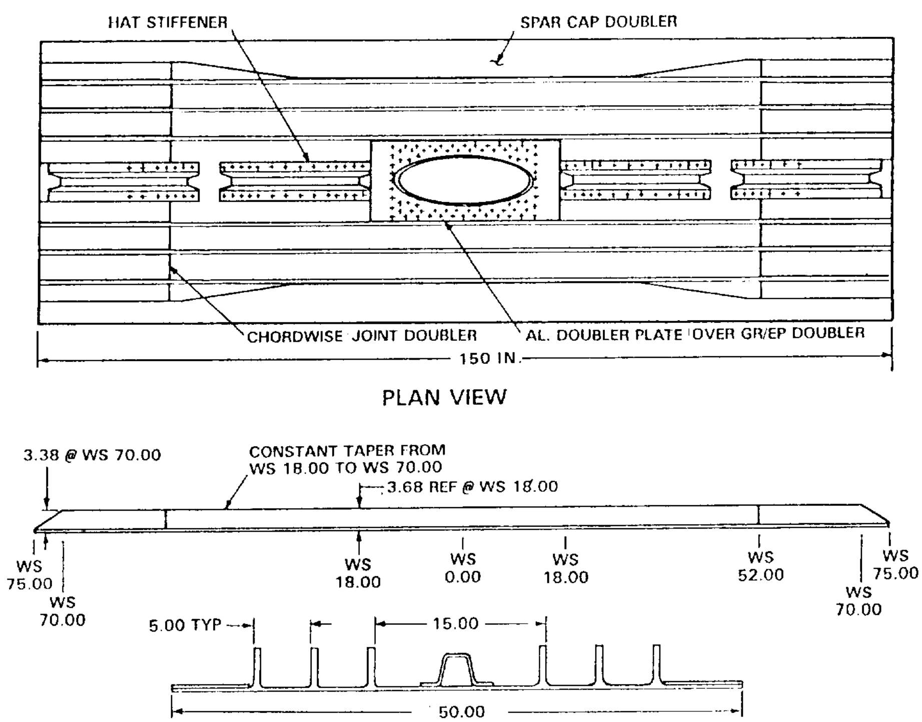


Figure 4. Upper Cover Box Beam Design

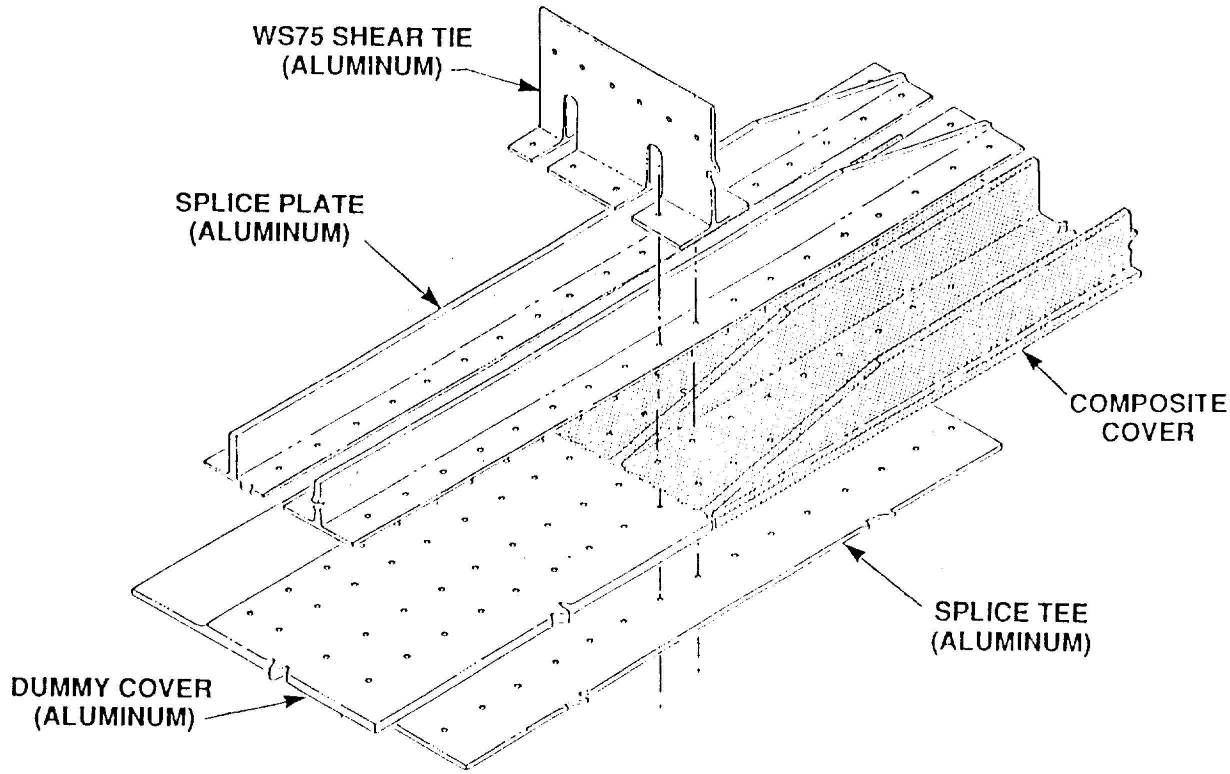
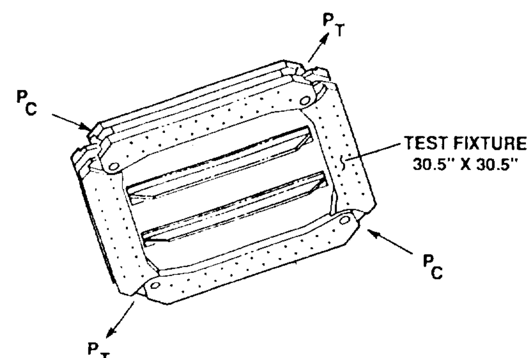


Figure 5. Cover Chordwise Splice

MATERIAL	MANUFACTURING METHOD	SHEAR FAILURE LOAD / AREAL-WEIGHT (IN.) ①
AS4/1806 FABRIC	HAND LAY-UP	410,900
AS4/1806 TOWPREG	FILAMENT WOUND	483,000
AS4/APC-2 TAPE	ROLL FORMED AND AUTO-CLAVE CONSOLIDATED	412,500
AS4/1806 FABRIC	HAND LAY-UP (POST BUCKLED DESIGN)	355,600

① 100FT-LB IMPACT



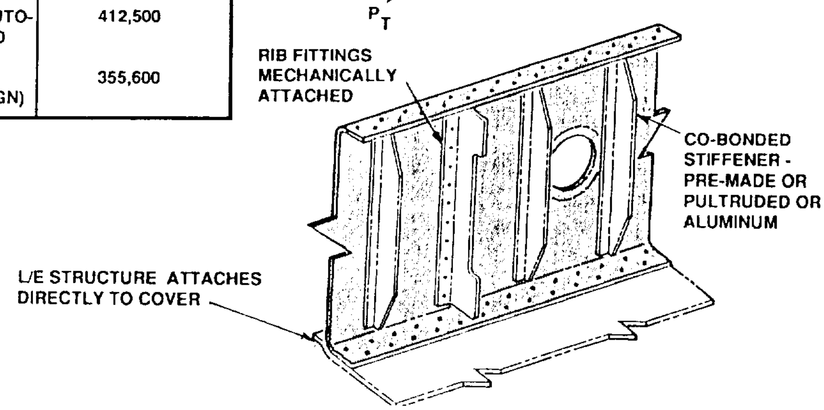


Figure 6. Wing Spar Designs

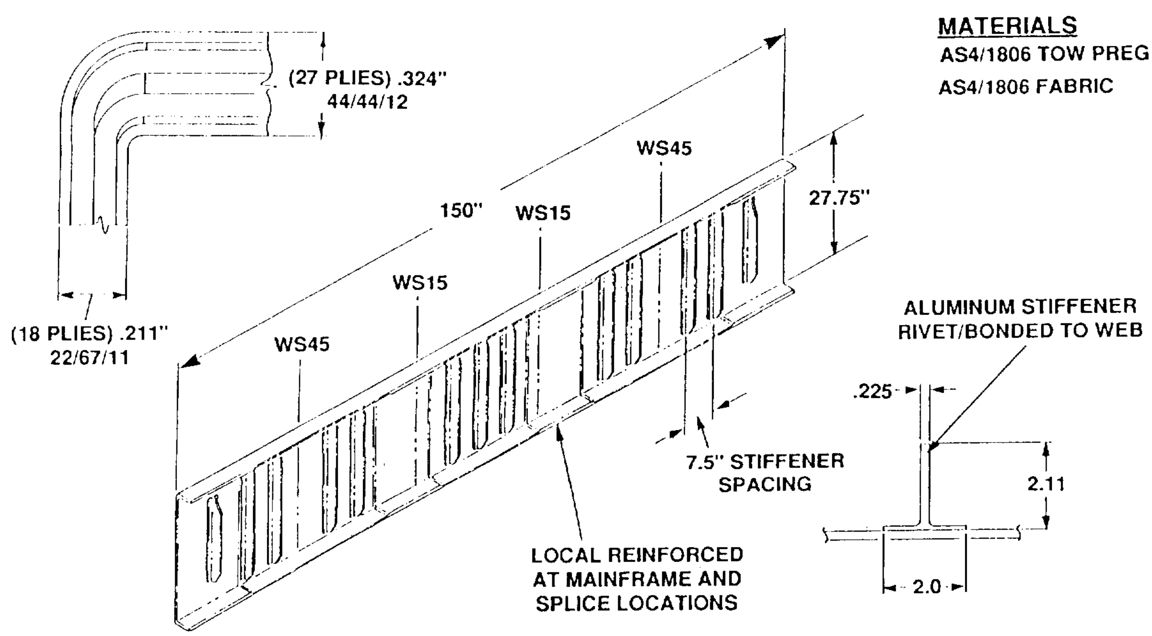


Figure 7. Spar Assembly

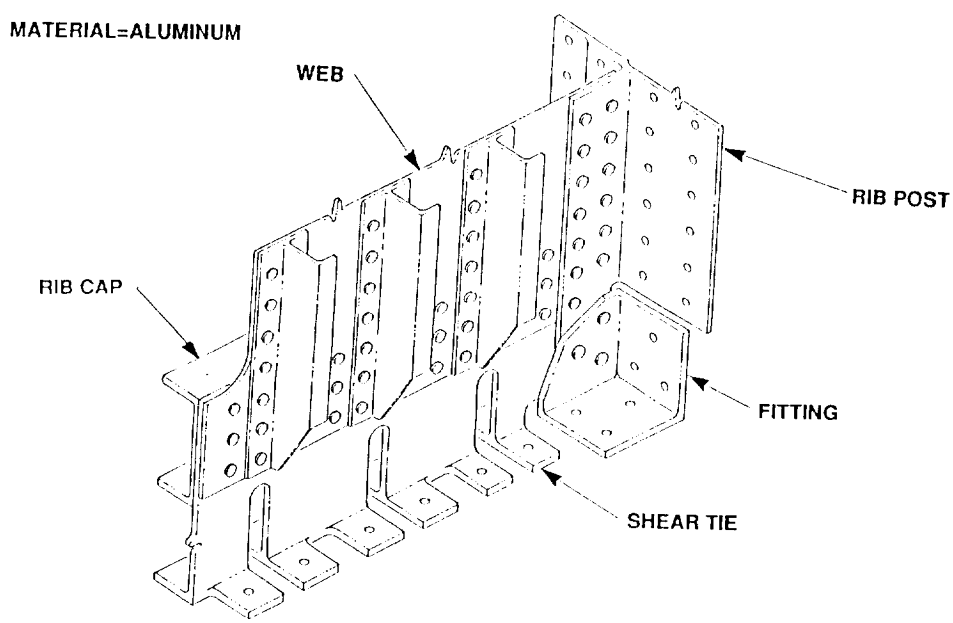


Figure 8. WS75 Rib

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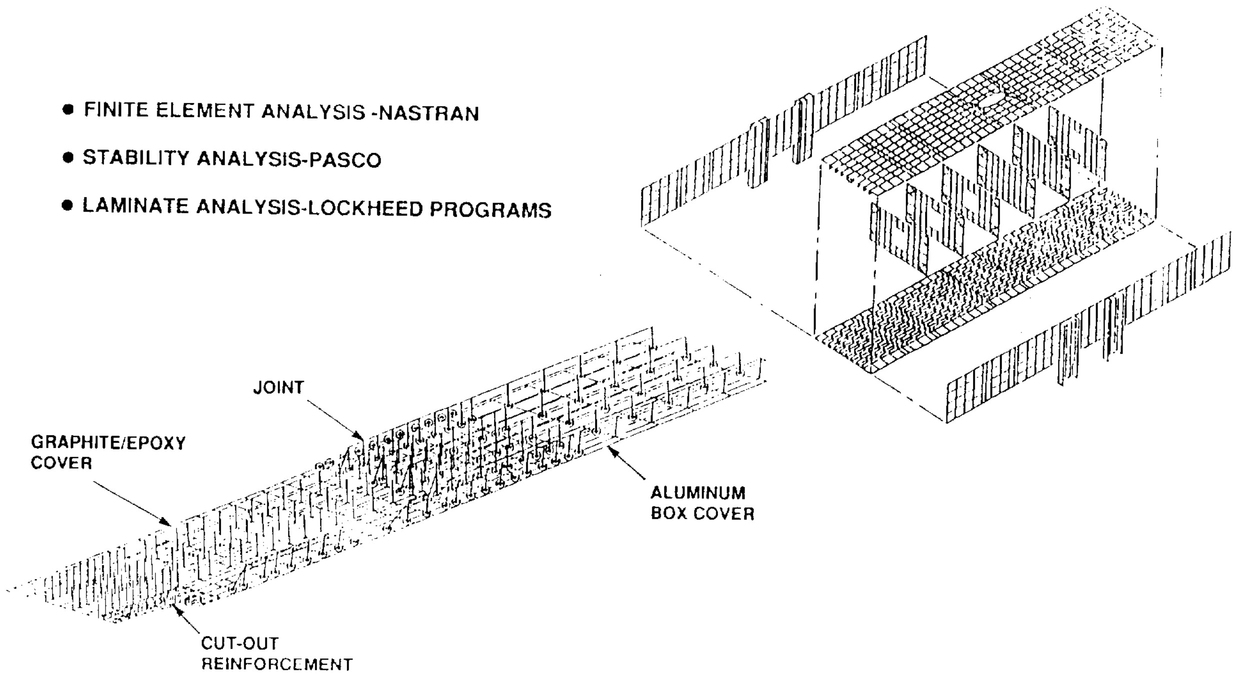


Figure 9. Structural Analysis Methods

Figure 10 presents the typical design allowables obtained for the AS4/1806 and AS4/974 fabric prepreg materials. These allowables were computed based on laminated tests, and in the case of the impacted laminate allowables, stiffened panel tests. Note that the allowable strain is plotted as a function of the percentage of plus/minus 45 degree plies within the laminate minus the percentage of 0 degree plies. This value is called the AML for angle minus longitudinal plies. For example, a quasi-isotropic laminate has an AML value of 25. The blade stiffener on the cover has an AML of -38 and the cover skin a value of 37.

Margins of safety were computed for numerous locations on the covers and spars using the applied strains and the design allowable strains. Several minimum margins of safety are presented in Figure 11. Both the upper cover and spar webs have a 0 margin of safety for the impact damaged condition. The lower cover and the spar cap are critical for bearing/bypass and net tension, respectively.

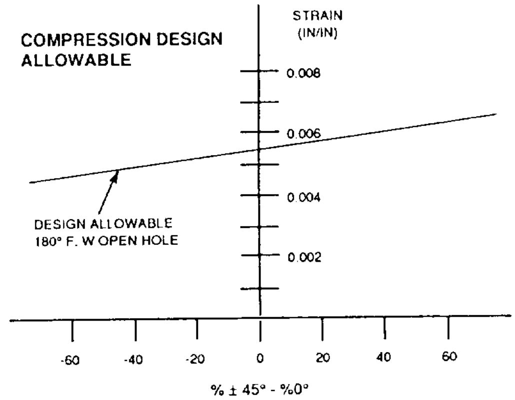
A preliminary design of a C-130 center wing box was completed using the design concepts and materials described for this technology integration box beam. Weights analysis indicated an overall savings of 25.4 percent compared to the metal baseline. The predicted spar weight savings was 35 percent and the cover was 28 percent. These weight savings could be improved if a higher modulus fiber such as IM7 were used in conjunction with the latest generation of toughened epoxies.

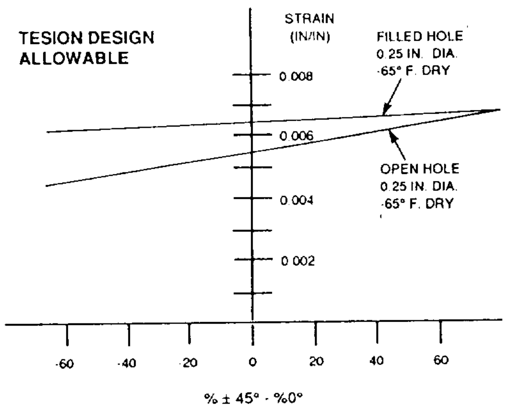
SPAR FABRICATION

The spars were fabricated by filament winding AS4/1806 towpreg onto a mandrel that when trimmed apart lengthwise produced both the front and rear spars at the

PLY LEVEL ELASTIC CONSTANTS

PROPERTY	UNIDIRECTIONAL FABRIC	BIDIRECTIONAL FABRIC
0° TENSILE MODULUS (MSI)	17.70	9.70
90° TENSILE MODULUS (MSI)	1.47	8.80
0° COMPRESSION MODULUS (MSI)	17.00	9.70
0° INPLANE SHEAR MODULUS (MSI)	0.62	0.62
0° POISSONS RATIO	0.30	0.05





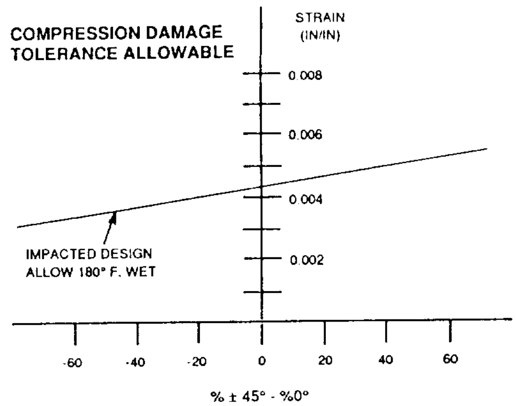
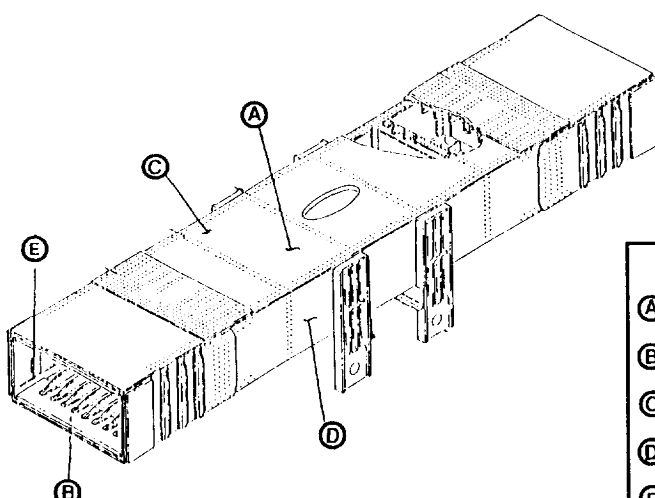


Figure 10. Design Allowables



LOCATION	M.S.	MODE
A UPPER COVER	0.00	CAI
B LOWER COVER	0.01	BRG/BYP
C UPPER COVER	0.04	BRG/BYP
D SPAR WEB	0.00	CAI
E SPAR CAP	0.05	NET TENSION

Figure 11. Minimum Margins of Safety

same time. Figure 12 shows the spars as filament wound and cured on the mandrel prior to removal. Figure 13 shows the two spars after separation. The aluminum stiffeners were fabricated, located on the spars, and drilled for fastener installation. The stiffeners were then phosphoric-acid anodized for bonding, adhesive coated with Hysol #9339 glass-microballoon filled adhesive and bolted and bonded in place. Figure 14 shows a spar with the stiffeners installed.

COVER FABRICATION

The covers were fabricated by separately laying-up C-channels, 64 tow stuffers and the skin laminate and then assembling the lay-ups for cure as shown in Figure 15. The C-channels were alternately laid-up on 'hard' mandrels and 'soft' mandrels to achieve both positive location of each blade stiffener and full fluid bag pressure during cure.

Note: 'Hard' mandrels were hollow aluminum mandrels formed into closed boxes by welding two Ls together. 'Soft' mandrels were U-shaped silicone rubber mandrels reinforced with included graphite fabric placed directly into the C-channel lay-ups to apply fluid pressure to the laminate while maintaining some stiffness for dimensional and shape control.

After laying up the skin laminate directly on the steel cover tool plate the center hard mandrel with its C-channel in place was positioned and pinned in place. (NOTE: Each hard mandrel, being aluminum to reduce worker handling weight, has a large difference in thermal expansion from the steel base; therefore, tooling pin holes were solid on one end and slotted on the other.) Next, a towpreg stuffer made of 64 tows was installed in the radius of the C-channel-to-skin joint. Then a soft mandrel C-channel layup was installed, another stuffer, another hard mandrel C-channel, etc., until the lower cover layup assembly was complete. Figures 16, 17, 18, and 19 show sequentially this layup process. A layer of FM 300 0.030 psf adhesive film is used between each layup interface. Thermocouples were installed, the tool corners padded, breather material applied, and the tool covered with the curing blanket which was sealed to the tooling base. The cover was then cured in the autoclave with 85 psi at 350°F for two hours. As seen in Figure 20, taken after unbagging and removal from the tool, the cover exhibited a curve which is due to differential shrinkage of resin at each blade stiffener location. This curve, however, is easily removed with moderate force at time of assembly with spars and ribs.

The cover has approximately four inches of trim excess on one end and 10 inches of excess on the opposite end which contains NDI standard flaws. This end will become the NDI standard for these panels after trimming to net size.

Additional material has to be added in a separate layup and cure cycle to serve as doublers for the load introduction box joint at each end of the covers. At the same time, spar cap doublers, separately laid up and cured, will be bonded on using FM 300 0.030 psf adhesive film. The hat stiffeners are to be installed on the cover with fasteners and micro-balloon filled paste adhesive, as were the stiffeners on the spars, in a bolted/bonded joint. After trimming the cover to final size, a 3-ply fiberglass layer overwrap will be installed with a vacuum cure at 250°F on the upstanding leg of each stiffener for damage tolerance protection as shown in Figure 2.

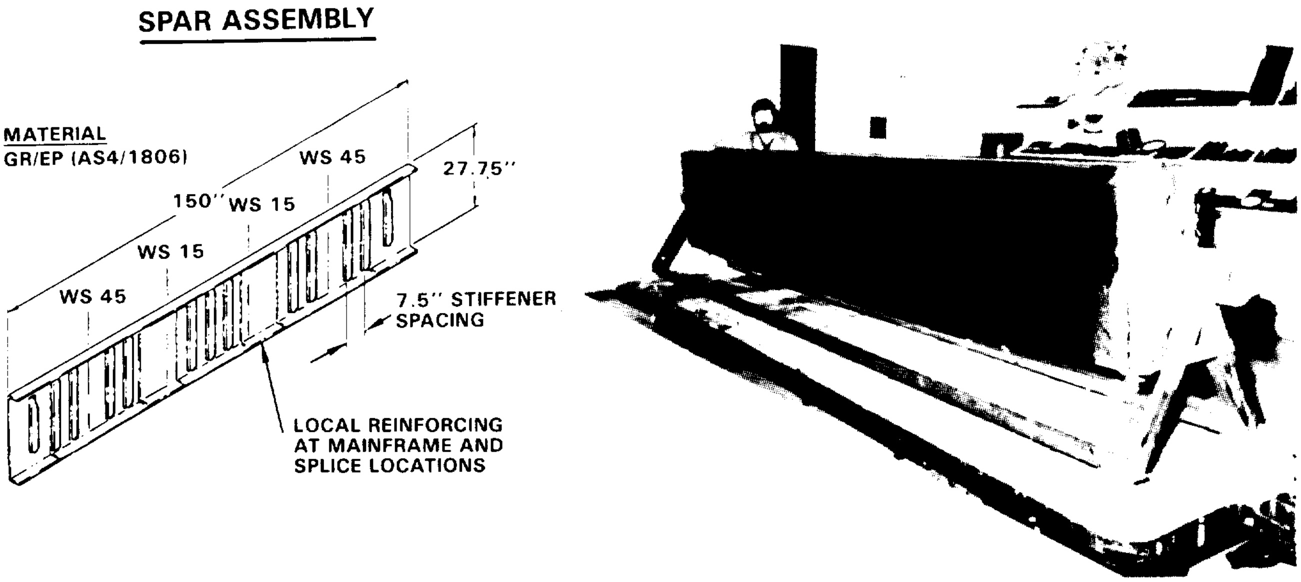


Figure 12. Spar Assembly

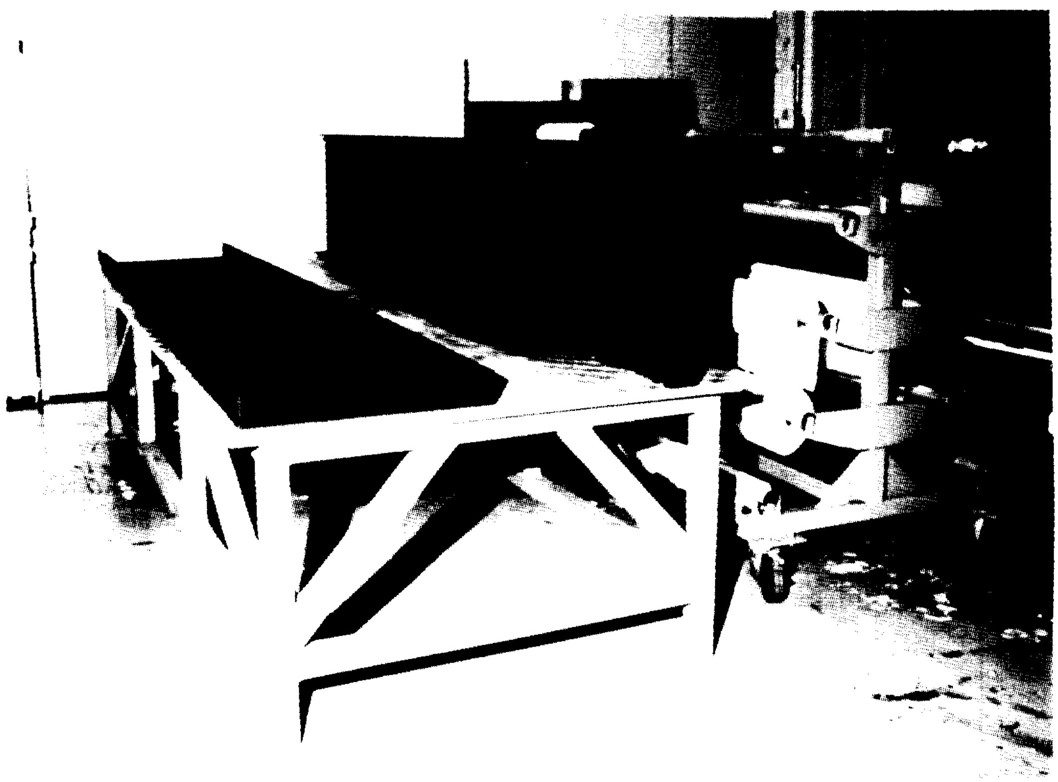


Figure 13. Two Spars

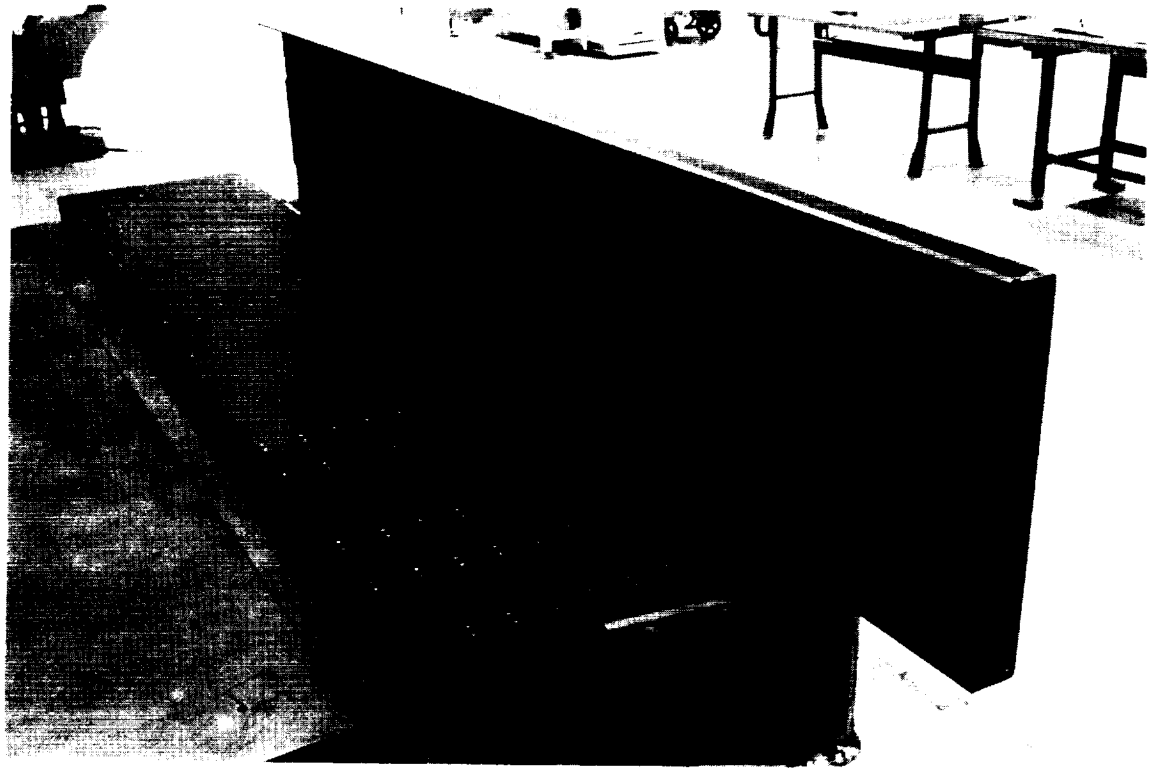


Figure 14. Spar Assembly

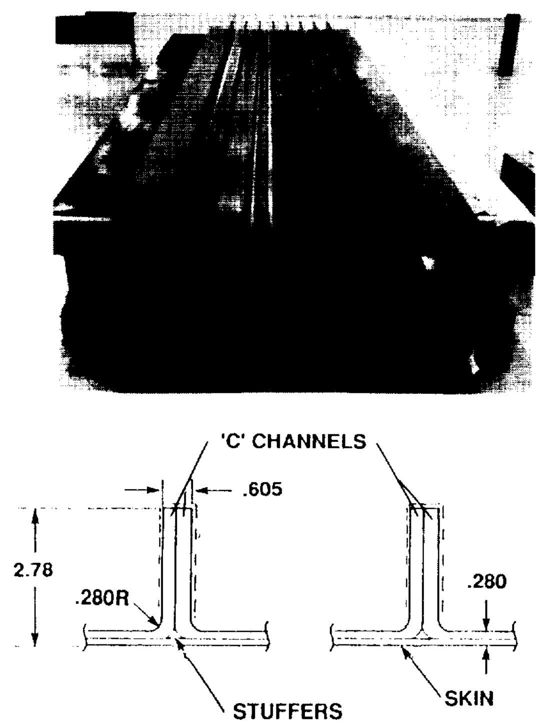


Figure 15. Skin Panel Layup Scheme

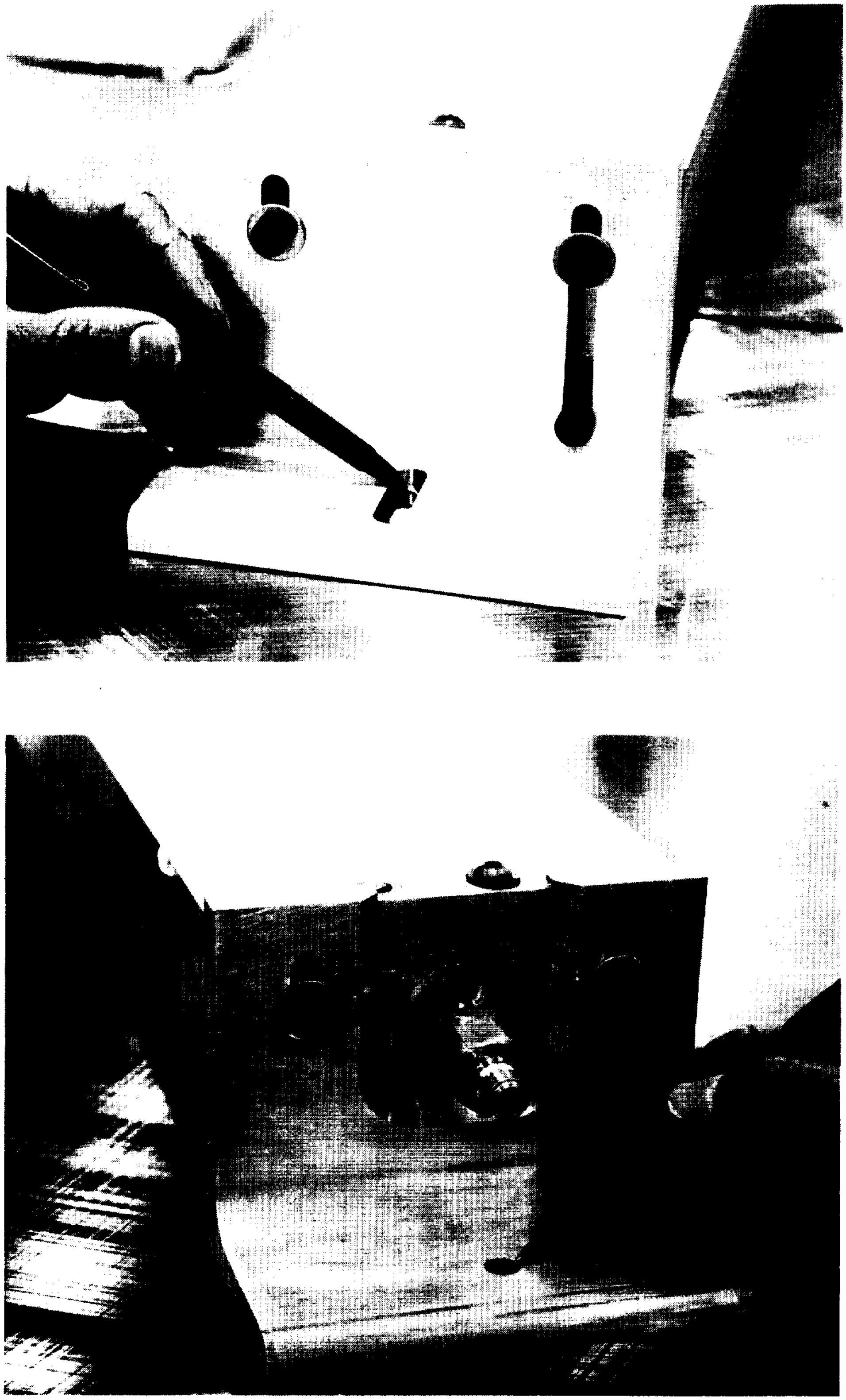


Figure 16. Solid versus Slotted Mandrel Tooling Holes

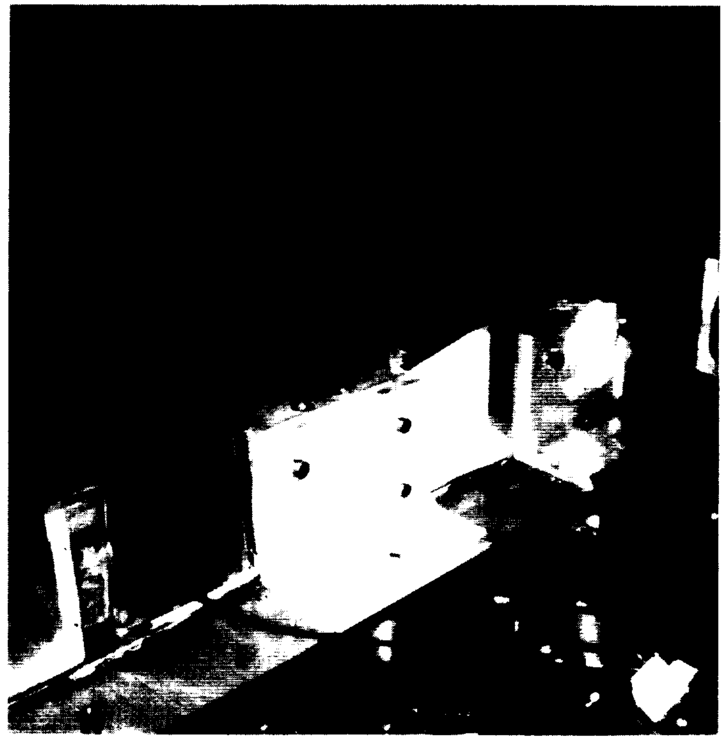


Figure 17. Soft Mandrel to be Inserted

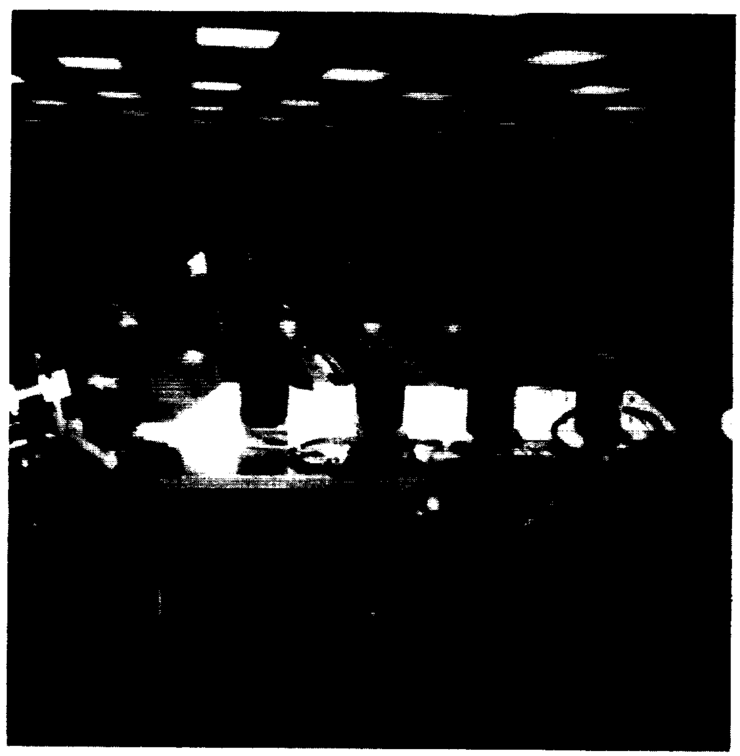


Figure 18. Panel Ready for Bleeder

Figure 19. Bagged Lower Cover in Autoclave

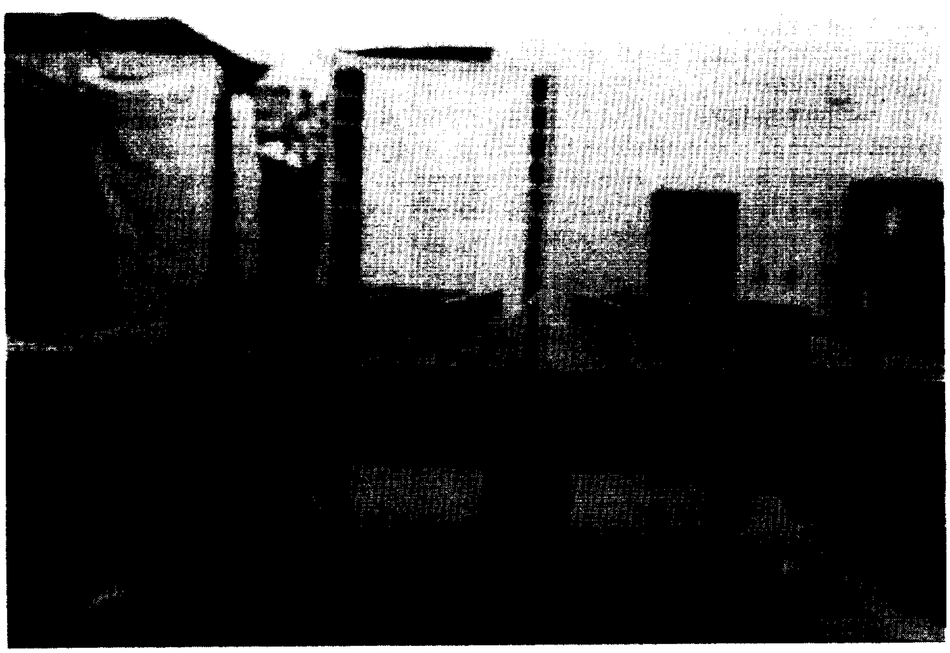


Figure 20. Curve in Panel Due to Blade Resin Shrinkage

RIB FABRICATION

The ribs being made of aluminum use standard aircraft assembly methods with mechanical fasteners as seen in Figure 21. An auto-fastener machine was used to install most of the rib assembly fasteners. Fastener locations were placed directly onto the part by using a mylar reproduction of the blueprint as an overlay.

BOX ASSEMBLY

The box assembly sequence will utilize assembly of the spar subassemblies with the rib subassemblies on a surface table using tooling knees initially for alignment and positioning. Once the spar-rib subassembly is assembled and squared, the covers will be joined to it and drilled using a Cybotec brand robot which will allow drilling of the cover-to-box holes without reaming. Although the box will not be sealed for fuel tightness, fasteners will be wet installed with corrosion inhibiting sealant where necessary to prevent corrosion between metal and composite surfaces. The box assembly includes an aluminum load introduction box extension on each end for testing. The test fixture is being fabricated under independent funding and will interface with the technology integration box beam via the aluminum box extension. Figure 22 details the assembly sequence of the box beam.

COST ANALYSIS

A detailed cost analysis compared a composite center wing structural box with an advanced aluminum version of the C-130 center wing box. The cost analysis evaluated the final technology integration box beam design, which was extrapolated to a full sized 80 by 440 inch wing box. The results demonstrate a potential 5 percent labor and material cost savings for a composite wing box compared to a new state-of-the art metallic design. Cost benefits are achievable in the current composite design concept through a reduction of labor costs; innovative design concepts result in less time required for fabrication and assembly operations. Also, automated manufacturing processes such as filament winding and pultruding have the potential to reduce costs. Estimated costs of the composite wing box, project recurring costs that will be incurred during a typical full-scale production program producing 200 ship sets of wing boxes. Figure 23 illustrates the cost breakdown for each major component as well as final assembly. Costs are distributed for both the advanced aluminum and composite wing box, illustrating relative costs of covers, ribs, spars, and assembly. Recurring production costs of the composite box are 95 percent of the baseline as a result of fewer parts in composite subassemblies and automated fabrication processes. Aluminum costs are based on actual C-130 cost history. Composite material costs are based on material vendors projections for material at high quantities. Fabrication costs, where possible, are generated from actual composite production experience. Where cost tracking data is not available, Value Engineering cost estimating methodology is used.

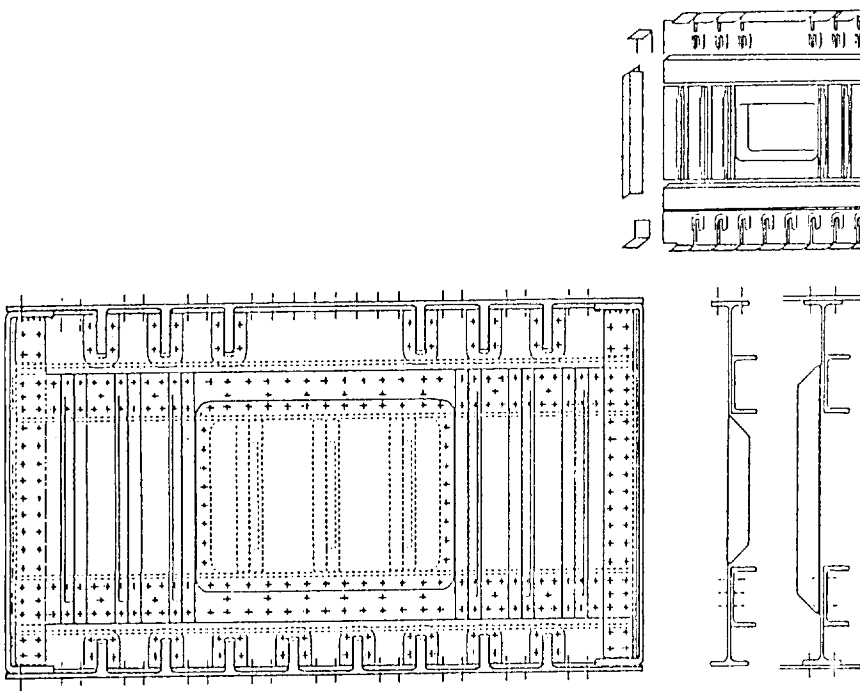


Figure 21. Rib Assembly

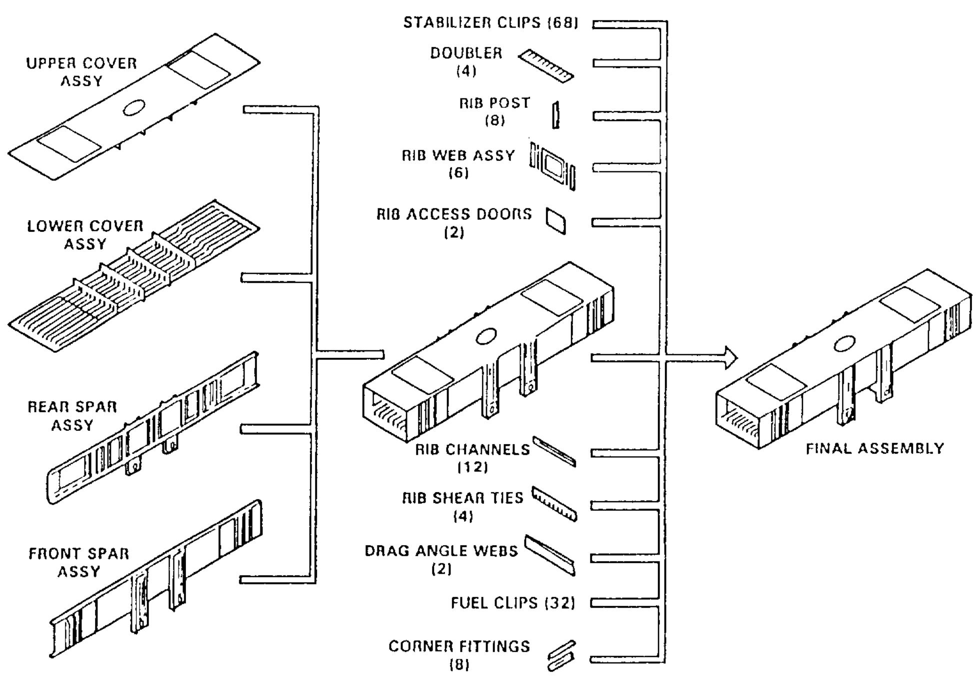


Figure 22. Task 5 - Box Beam Assembly Breakdown

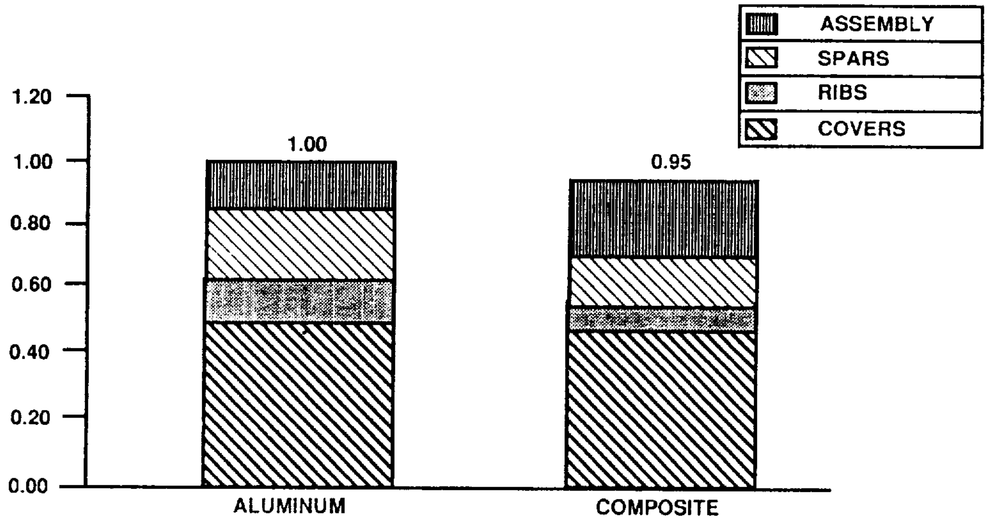


Figure 23. Wing Box Relative Cost Comparison Cost Breakdown

The covers comprise 49 percent of the total cost and account for 64 percent of the total weight of the wing box (reference Figure 24). A meaningful cost reduction could most easily be achieved through reducing costs associated with the covers. Assembly costs also have a considerable impact on costs, accounting for 25 percent of the total. Cost reductions for the spars and ribs will not result in an appreciable cost savings since their impact on total costs is only 10 percent and 16 percent, respectively.

Figure 25 further illustrates the total cost breakdown, showing material and labor costs separately. This provides a more specific means with which to target and assess cost drivers. Material costs for the covers stand out, accounting for 35 percent of the cost. Figure 26 shows a breakdown of material costs, demonstrating the significance of the cost of the covers as a percentage of total material costs. Approximately 74 percent of the total material costs is in the covers as shown. Material costs are based on projected costs estimated by vendors for graphite/epoxy prepreg at 10,000 pounds per year quantities. The plus/minus and minus/plus 45 degree knitted fabric is priced almost 87 percent higher than the uni-directional and 0/90 degree fabric. There is a potential for reduction in the former as the vendors have limited experience producing this material; consequently, their estimate may be conservative. A reduction in this price could have a significant effect on cover material costs as the plus/minus 45 and minus/plus 45 degree fabric accounts for 47 percent of the cover material costs.

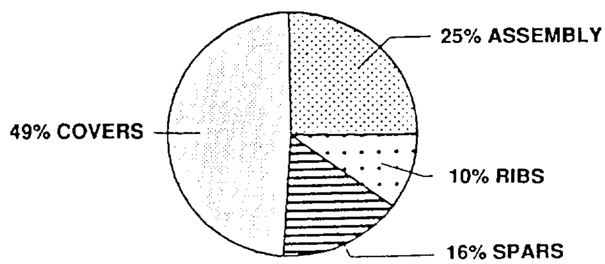


Figure 24. Total Cost Breakdown

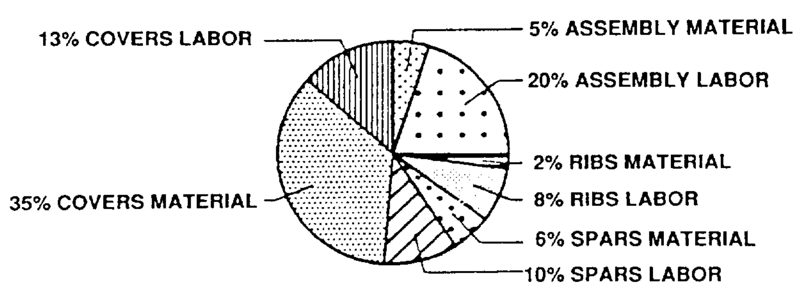


Figure 25. Labor Cost Breakdown

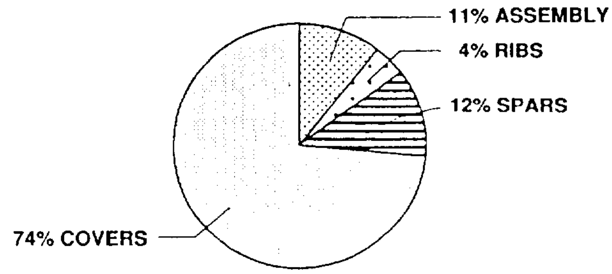


Figure 26. Material Cost Breakdown

Assembly costs are 25 percent of the total cost, 20 percent of which is assembly labor and 5 percent material. Assembly labor costs account for 39 percent of the total labor cost (reference Figure 27); assembly is labor intensive due to the necessity of a two-step drilling procedure required for graphite/epoxy composites to obtain acceptable holes. Improved composite drilling techniques/equipment or a reduction in the overall part/number of fasteners (see Figure 28) could significantly reduce the total costs. Material costs, at 5 percent of the total, are not a driver even though titanium fasteners are required.

Spar costs account for 16 percent of the total cost and probably represent an optimized design; the cost is based on a filament wound spar which significantly reduces the number of parts and fasteners, compared to a new metallic spar design. Further cost reduction is unlikely and would have a negligible effect on the total cost.

The ribs are only 10 percent of the total costs, 8 percent of that for labor. It is assumed that the skins and channels would be hand laid-up. Material costs are based on the same assumptions as the covers, and may be reduced; however, only 4 percent of the total material costs (reference Figure 26) is for the ribs. This is not an ideal target for emphasis on cost reductions.

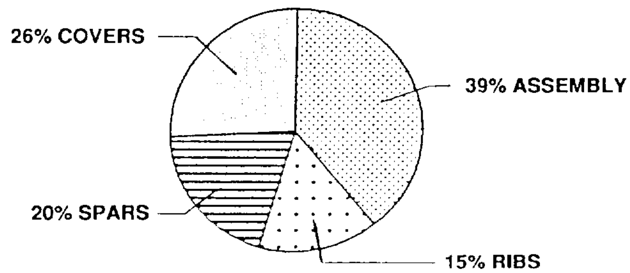


Figure 27. Labor Cost Breakdown

CONSTRUCTION TYPE	PARTS	FASTENERS
ALUMINUM "C130 CENTER WING" "FRONT SPAR"	160	3500
COMPOSITE SPAR	40	200

Figure 28. Part/Fastener Comparison

CONCLUDING REMARKS

The concurrent engineering approach used in this project has resulted in a wing box design which has a 25 percent weight saving and a 5 percent cost saving compared to the baseline advanced metal wing box. Incorporation of improved materials and the evaluation of alternatives to the bias fabrics could lead to further reductions in weight and acquisition costs. Spars were successfully filament wound, back-to-back on a common mandrel. Box covers were also successfully co-cured. These successful fabrication demonstrations point up still more lower-cost fabrication methods that could be incorporated in the future.