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NASA Contractor Report 172330

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Development of Laminar Flow Control Wing Surface Composite Structures

L. B. Lineberger, et al.

LOCKHEED-GEORGIA COMPANY
Marietta, Georgia 30063

CONTRACT NAS 1-17487
May 1984

(NASA-CR-172330) DEVELOPMENT OF LAMINAR FLOW CONTROL WING SURFACE COMPOSITE STRUCTURES Final Report (Lockheed-Georgia Co.) 135 p Avail: NTIS HC A07/MF A01 CSCL 01A G3/01

N87-21849

Unclas 0072557

National Aeronautics and Space Administration

Langley Research Center
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be three (3) years from date indicated on the document.

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Space Administration

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FOREWORD

Contract NAS1-17487 between the National Aeronautics and Space Administration and the Lockheed-Georgia Company, effective August 8, 1983, provides under Task No. 1 for preparing a final NASA Contractor Report, documenting the NAS1-16235 LFC Laminar-Flow Control Wing Panel Structural Design and Development (WSSD); Design, Manufacturing, and Testing Activities for the period of September 1, 1980, through December 23, 1981. Contract NAS1-17487 is sponsored by the Aircraft Energy Efficiency (ACEE) Project Office of the Langley Research Center, with Dal V. Maddalon serving as Technical Monitor. This document, submitted in fulfillment of DRL-003-1 of the subject contract, constitutes the final report.

At the Lockheed-Georgia Company, the Contract was accomplished under the cognizance of R. H. Lange, Manager, Advanced Concepts Department, with L. B. Lineberger serving as Project Manager. Principal participants in this contract effort were as follows:

R. T. Beall	Manufacturing Development
G. J. Gilbert	Development Testing

Contract NAS1-16235 (WSSD) was sponsored by the Aircraft Energy Efficiency Project Office of the Langley Research Center, with Jack Cheely serving as NASA Technical Monitor. At the Lockheed-Georgia Company, the WSSD contract was accomplished under the cognizance of R. H. Lange, Manager, Advanced Concepts Department, and R. F. Sturgeon, LFC Program Manager, with R. R. Eudaily serving as Project manager. Principal participants in WSSD contract effort were as follows:

J. A. Bennett	Aerodynamics
L. B. Brandt	Aerodynamics
S. D. Higham	Wing Design
L. B. Lineberger	Structure Development
Tom Dasher	Structure Development
J. A. Kizer	Structure Development/Test
R. D. O'Brien	Production Costs
R. T. Beall	Manufacturing Development
L. L. Allen	Quality Assurance
T. D. Stultz	Maintainability
G. J. Gilbert	Testing
R. S. Ferrill	LFC Systems
J. G. Tibbetts	LFC Systems

This report is identified as LG84ER0035 by Lockheed-Georgia Company for internal control purposes.

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1.0 SUMMARY

This report documents Lockheed's results in designing, fabricating, and testing in the September 1980 - December 1981 period under NASA Contract NAS1-16235, "Laminar Flow Control Wing Panel Structural Design and Development," (WSSD) which continues the development of technology for the integrated LFC wing structural concept identified in Phase I, Reference 1. The ultimate objective is to permit incorporation of LFC into long-range commercial jet transports in the post--1990 period. The specific objectives of the WSSD project reported herein support this program objective and include; continue development of integrated LFC wing structural concept identified in Phase I, and demonstrate that this concept can be efficiently applied to surface of a future commercial transport.

In satisfying these objectives, the project was organized into major tasks.

1.0 Conduct a preliminary design of the wing of a 1993 LFC commercial transport.

2.0 Design a surface panel for the selected LFC wing and verify by ancillary tests.

3.0 Develop manufacturing processes for the LFC surface panel and estimate costs for manufacture and maintenance of LFC wing surface.

4.0 Verify concept by fabricating the ancillary and demonstration panels.

An in-depth preliminary design of the baseline LFC wing was accomplished during Task 1.0 of the WSSD project. Structural members were located and sized. The LFC suction surfaces and internal ducting were also located and sized. The wing design included the following tasks:

- LFC wing defined during Phase I was updated and used for the baseline LFC wing

- Structural loads and stiffnesses were calculated for use in preliminary design of baseline wing

- The LFC suction surfaces and internal ducting requirements were defined

- An in-depth preliminary design of the LFC wing structure and LFC systems was conducted which included

  - Design loads, stresses and strains

  - Selection processes for alternate concepts

  - Criteria for surface tolerances

  - Consideration of environmental effects

- Accessibility and replaceability considered in designs

- Materials

- Integration of LFC systems

Detail design and verification of the surface panel was accomplished by Task 2.0. This task covers the detail design analysis, testing and verification of the integrated LFC structural concept for the surface panel. The surface panel design included the tasks:

o Detail design of surface panel

  - Ancillary test plans for all Material Verification (MV) & Concept Selection (CS) specimens

  - Details of surface panel being developed by MV & CS specimens

o Selected materials

o Verified material selections

o Details of all concept selection specimens defined for manufacture of specimens

o Detail design of concept verification and concept demonstration panel

o Preliminary plans made for testing concept verification and demonstration panels

NOTE: The concept verification and concept demonstration large panels were not fabricated or evaluated under the original WSSD program.

Manufacturing Development, Task 3.0, is the major task in the WSSD project. This task covers the development of manufacturing processes, estimated costs and maintenance of the LFC panel. Problems were encountered in processing material for the material verification panels, but these were solved. The concept selection specimens were more complex than planned, and extra effort was required to develop the manufacturing processes and tool concepts for these specimens.

o Conducted producibility design studies in conjunction with concept selection process during Task 1.0

o Manufacturing processes were developed for surface panel

  - All material verification specimens were fabricated

  - Manufacturing procedures for concept selection specimens were developed

- Tooling procedures for surface panel were developed
  - Tooling details were developed for critical details by fabricating of CS specimens
  - Tool for surface panel were designed
- Design manufacturing interfaces have controlled design selections
- Inspection, maintenance and repair procedures were evaluated for the surface tolerance criteria
- Cost of the baseline LFC airplane was estimated and compared to non-LFC airplane. Fuel costs are shown to be approximately $4,000,000 per year lower for the LFC aircraft. The calculation shows that the lower fuel costs for LFC offset the higher incremental costs of LFC in less than six months. The mission fuel weight was 21.7 percent lower for the LFC aircraft. The empty weight for the LFC aircraft was only 0.6 percent higher. This results from the efficiency of the integral-with-structure suction system which imposes a penalty of just 0.71 lb/ft^2. From these data, it can be seen that the development effort expended during the contract, continued the design, development, and tests of the highly efficient LFC wing box structure.

Planning for Task 4.0, Fabrication of Demonstration Panel was completed.

2.0 INTRODUCTION

The recognition of potential long-term shortages of petroleum-based fuel, evidenced by dramatic increases in costs and periods of limited availability since 1973, emphasized the need for improving the fuel efficiency of long-range transport aircraft. In 1976, in response to this need, the NASA established the Aircraft Energy Efficiency (ACEE) program with the objective of maintaining the U.S. competitive advantage through development of new technology for fuel efficiency. Of all advanced-technology concepts currently under consideration for application during the next two decades, Laminar Flow Control (LFC) offers the greatest potential for improving the fuel efficiency of transport aircraft. Consequently, LFC is included as one element of the ACEE program.

Both the theoretical methods and engineering and design techniques requisite to the application of LFC have been reasonably well-known since the mid-1940's. The validity of this background and the potential of LFC were partially evaluated in the 1960-1966 period by Northrop as a part of the X21A LFC Demonstration Program.

In the process of formulating the current LFC Program, the NASA sponsored a "Workshop on Laminar Flow Control," at the NASA Langley Research Center in April 1976. Attendees included representatives of the aircraft industry, the airlines, the Department of Defense, and the NASA. It was the general concensus of the participants in this workshop that the following tasks must be accomplished prior to the incorporation of LFC on an operational aircraft:

(1) The development of LFC structure and systems with acceptable weight and cost penalties.

(2) The development of procedures for the economical manufacturing of LFC structure in a production environment.

(3) Demonstration of the operational reliability of LFC in the airline environment.

NASA formulated a three-phase program with the goal of developing LFC technology to permit application to aircraft in the 1990 period.

The Phase I effort, concluded in September 1978, resulted in the definition of candidate LFC systems for application to future production aircraft. Phase II, of which this contract is a part, involves the design and development of selected structural concepts, and initial development and testing of selected leading-edge subsystems. The final phase, Phase III, originally envisioned to encompass the design, fabrication, and flight demonstration of an integrated LFC system in a validator aircraft, will be redefined at some future time.

A central problem in the definition of a feasible production configuration for LFC transports is the development of LFC surface designs which satisfy aerodynamic requirements without imposing unacceptable structural weight penalties, manufacturing costs, and operational requirements. Consequently, during Phase I of the LFC program, extensive investigations were conducted in the development of structural concepts for the wing-box region of the wing of an LFC transport. As a part of the development, alternative structural concepts were evaluated, detailed designs were developed for selected concepts, manufacturing procedures were established, and full-scale structural specimens were fabricated and tested.

The selected LFC surface design for the wing-box region is a structural skin and hat-section stiffener configuration with LFC ducting and metering integrated into the structure. The structural elements are fabricated of graphite/epoxy composites, with a titanium outer face sheet for lightning protection and resistance to erosion and corrosion. During Phase I, three 3 ft x 5 ft LFC surface panels were fabricated and subjected to extensive environmental and structural testing which validated the design concept.

The "Laminar Flow Control Wing Panel Structural Design and Development" (WSSD) project continues the development of the integrated LFC wing structural concept identified during Phase I of the LFC Program sponsored by the ACEE Project office of the Langley Research Center. This report summarizes progress in the application of this concept to the wing of a 1993 LFC transport. Details of the LFC system and wing surface structure were developed by a preliminary design of the wing and verified by an ancillary test program. Costs of the LFC transport were compared to those of an equivalent technology non-LFC transport designed for the same mission. Manufacturing processes were described and plans were outlined for the fabrication and testing of a large section of the wing surface to demonstrate that the integrated LFC wing surface structural concept can be efficiently applied to a future commercial transport.

3.0 ABBREVIATIONS AND SYMBOLS

ACEE	AIRCRAFT ENERGY EFFICIENCY
ACT	ACTUATOR
AE	ACOUSTIC EMISSION
AL/AL	ALUMINUM/ALUMINUM
ANSWER	ANALYTIC STRUCTURAL WEIGHT ESTIMATION ROUTINE
ARF	AUTOMATED RESIZING FOR FLUTTER
ASSY	ASSEMBLY
AUX	AUXILIARY
C	CHORD, CRUISE
CKS	COUNTERSINK
CL	CENTERLINE
CO	COMPANY
COMM'L	COMMERCIAL
CONT	CONTROL
CONT'D	CONTINUED
CS	CONCEPT SELECTION
CV	CONCEPT VERIFICATION
D	DIVE, DIAMETER
DELAM	DELAMINATION
DEMO	DEMONSTRATION
DIA	DIAMETER
DRL	DOCUMENTS REQUIREMENT LIST
EI	BENDING STIFFENESS
EQUIV	EQUIVALENT
EQ	EQUAL
FAR 25	FEDERAL AVIATION REGULATION-TRANSPORT CATEGORY
FG	FIBERGLASS
FLT	FLIGHT
FT	FEET
FT/SEC	FEET PER SECOND
FTG	FITTING
g	GRAVITATIONAL CONSTANT, GRAMS
g/m^2	GRAMS PER METER SQUARED
gal	GALLON
GJ	TORSION STIFFNESS
GR/EP, G/E	GRAPHITE EPOXY
IN	INCHES
INBD	INBOARD
INDUST	INDUSTRY
J	JAY
K	KIPS, 1000 POUNDS
KEAS	KNOTS EQUIVALENT AIRSPEED

KIPS/IN, K/IN	1000 POUNDS PER INCH
KSI	1000 POUNDS PER SQUARE INCH
LB	POUNDS
LB/FT^2	POUNDS PER SQUARE FOOT
LE	LEADING EDGE
LFC	LAMINAR FLOW CONTROL
M	MACH NUMBER
MAC	MEAN AERODYNAMIC CHORD
MAX 100X	MAGNIFY 100 TIMES
MFG	MANUFACTURE
MI	MILE
MIN	MINUTE
MV	MATERIAL VERIFICATION
N	NAUTICAL, LOADING
NASA	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
NDI	NON DESTRUCTIVE INSPECTION
NO	NUMBER
P	LOAD, POUNDS
POR	POROSITY
PSI	POUNDS PER SQUARE INCH
q	SHEAR LOAD
R	RANK, STRESS RATIO, RADIUS
RC	RESIN CONTENT
RDT&E	RESEARCH, DEVELOPMENT, TEST, AND ENGINEERING
REF	REFERENCE
RMU	RIPPLE MEASUREMENT UNIT
RT	ROOM TEMPERATURE
S	SCORE, SYMMETRY
SQ	SQUARE
STA	STATION
t	THICKNESS
TBD	TO BE DETERMINED
T-BAR, $\bar{t}$	AVERAGE THICKNESS INCLUDING SKIN PLUS STIFFENER
TEMP	TEMPERATURE
TI	TITANIUM
TYP	TYPICAL
U_{DE}	GUST VELOCITY
WRP	WING REFERENCE PLANE
WS	WING STATION
WSSD	LAMINAR-FLOW CONTROL WING PANEL STRUCTURAL DESIGN AND DEVELOPMENT

X	AIRCRAFT AXIS - AFT
X/C	CHORD LOCATION
X21A	NORTHROP LFC TEST AIRCRAFT (B-66)
Y	AIRCRAFT AXIS - OUTBOARD TO LEFT
Z	AIRCRAFT AXIS - UP
ZFW	ZERO FUEL WEIGHT

SYMBOLS

&	AND
"	INCH
%	PERCENT
η	WING SEMISPAN LOCATION RATIO (WS/SEMISPAN LENGTH)
σ	STRESS
ϵ	STRAIN
μ	MICRO
o	DEGREE
#	NUMBER
1st	FIRST
2nd	SECOND

4.0 WING CONFIGURATION, DESIGN CONCEPTS, AND MATERIALS

4.1 WING CONFIGURATION

During Phase I, Lockheed conducted a comprehensive system study to evaluate the advantages of Laminar Flow Control (LFC) for future transport aircraft in the 1985-1995 time period. The study showed the use of LFC resulted in significant reductions of aircraft weight, fuel consumption, and direct operating costs.

Investigations were conducted to determine the optimum configuration for a 400-passenger long-range transport featuring LFC. The LFC aircraft configuration was optimized for a 84,800 lb payload, a range of 6500 n mi at cruise M = 0.80 and 10,000 ft field length. This aircraft included advanced technology applications such as supercritical airfoil shapes, active controls, and composite primary and secondary structures (Reference 1).

4.1.1 1993 LFC Transport

The optimum configuration of the long-range 1993 transport with LFC is illustrated in Figure 1. Engines are mounted on pylons extending from the rear fuselage. This location provides a clean wing for the LFC suction system. The 240 ft long fuselage is sized to accommodate a typical 10/90 passenger mix with 40 in first class, seated 6 abreast, and 362 in tourist, seated 10 abreast. Space allowances are made for galleys, lavatories, closets, cabin crew provisions and rest areas for flight crews. Space for LD-3 cargo containers is provided in the underfloor area forward of the wing box and aft of the landing gear compartment. A bulk cargo bay is also provided at the rear of the pressurized belly. These cargo bays will accommodate 37,000 lb of cargo.

A "tee-tail" configuration is used with the rear mounted engines.

LFC suction capability is provided for the upper and lower surfaces of wing and horizontal stabilizer. An independently driven suction pump for the LFC system is located under each wing root.

4.1.2 Wing Dimensions 1993 LFC Transport

Figure 2 shows the planform geometry, basic dimensions, and airfoil cross section of the LFC wing. The wing is swept 25° at the leading edge and the semispan is 1486.68 in (123.89 ft). The wing chord tapers from 516.34 in at the root to 304.20 in at the bat break to 132.91 in at the tip. Wing area (batted) is 5724 square feet. Aspect ratio is 11.6, and the taper ratio is 0.35.

The airfoil section shown for the bat break is applicable, when scaled by chord length, from bat break to wing tip. Wing thickness ratio is 0.1128. Inboard of the bat break the airfoil tapers to the cross section shown at the root.

LFC suction capability is provided from the leading edge to 75 percent of the basic wing chord on both upper and lower surfaces.

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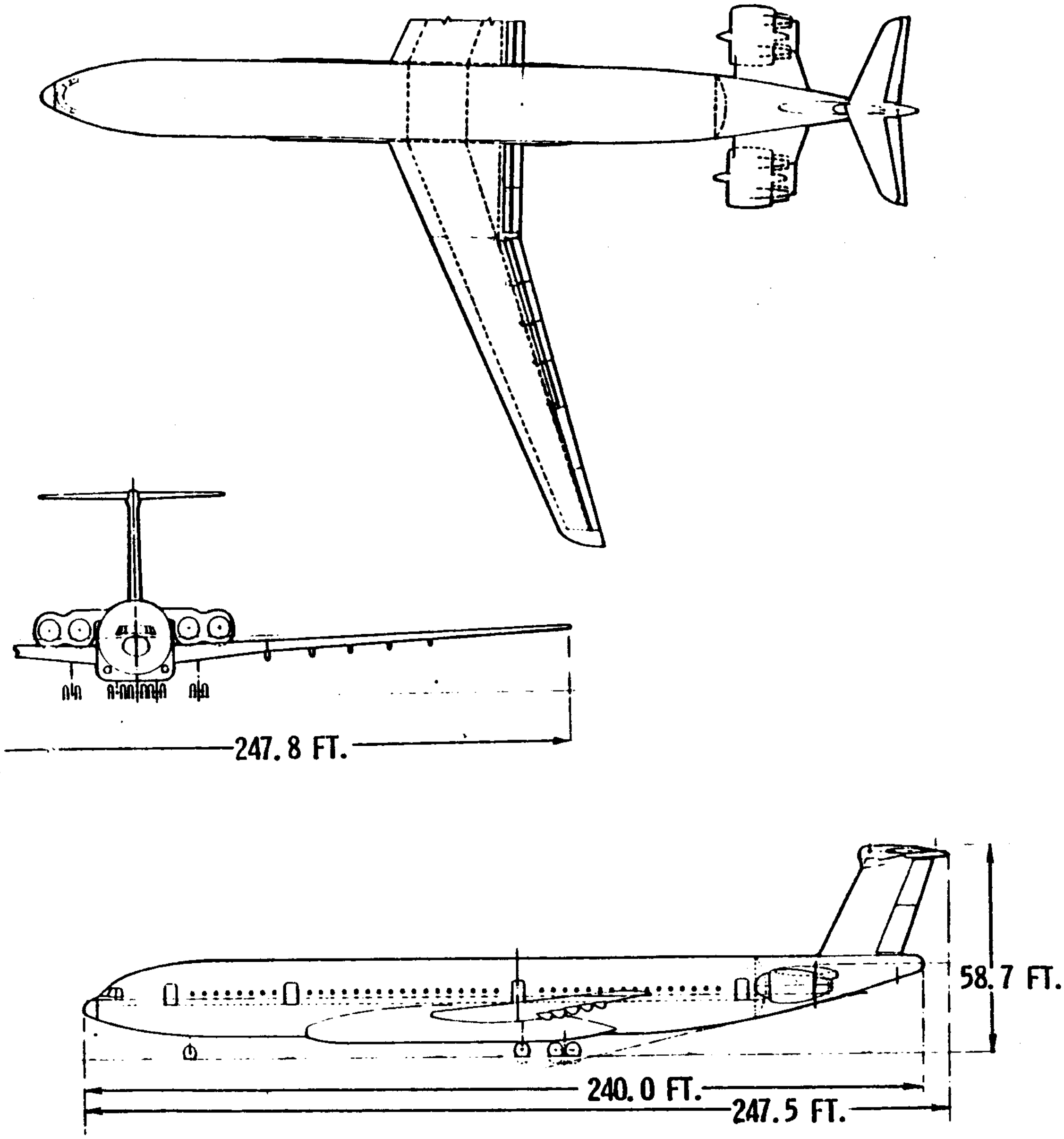

Figure 1. 1993 LFC Transport

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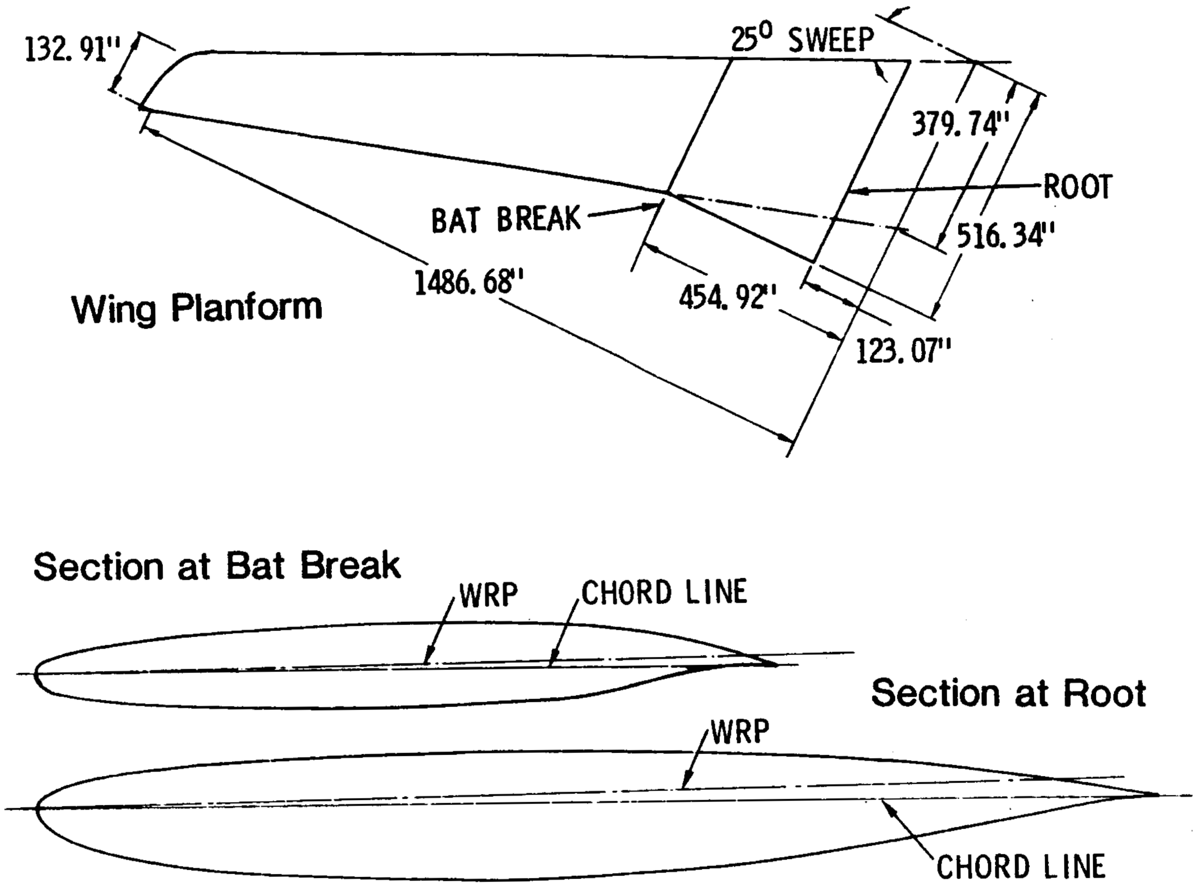

Figure 2. Wing Dimensions & Contours - 1993 LFC Transport

4.1.3 LFC Wing Structural Design

The baseline LFC wing concept developed during the LFC WSSD project is shown in Figure 3. Airflow from the slots in the leading edge, upper and lower surfaces through the internal system of ducts to the main trunk ducts in the leading edge is shown by the arrows.

Figure 4 shows the layouts of the basic, structural geometry of the wing and the location of the principal structural members.

The outer box structure extends from the root splice at butt line 123 to the wing tip. The wing spars are located at 18 percent and 75 percent of the basic wing chord. Ribs are spaced generally 35 in apart. Each alternate rib serves as a chordwise duct for the LFC system. The structure is spliced (for autoclave considerations) at the batted wing break (W.S. 454.9) and at a point approximately 2/3 semispan (Box Sta 1100). Ribs at the splices are bulkhead type and serve as fuel barriers. All other ribs are truss type.

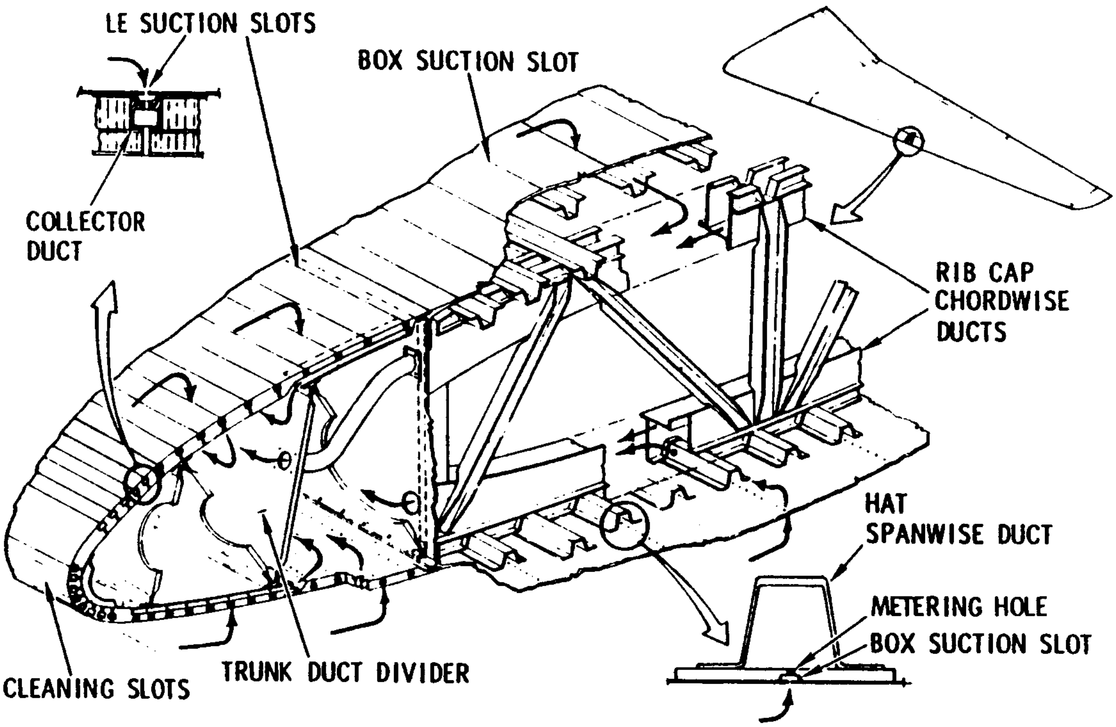

Figure 3. LFC Wing Structural Design

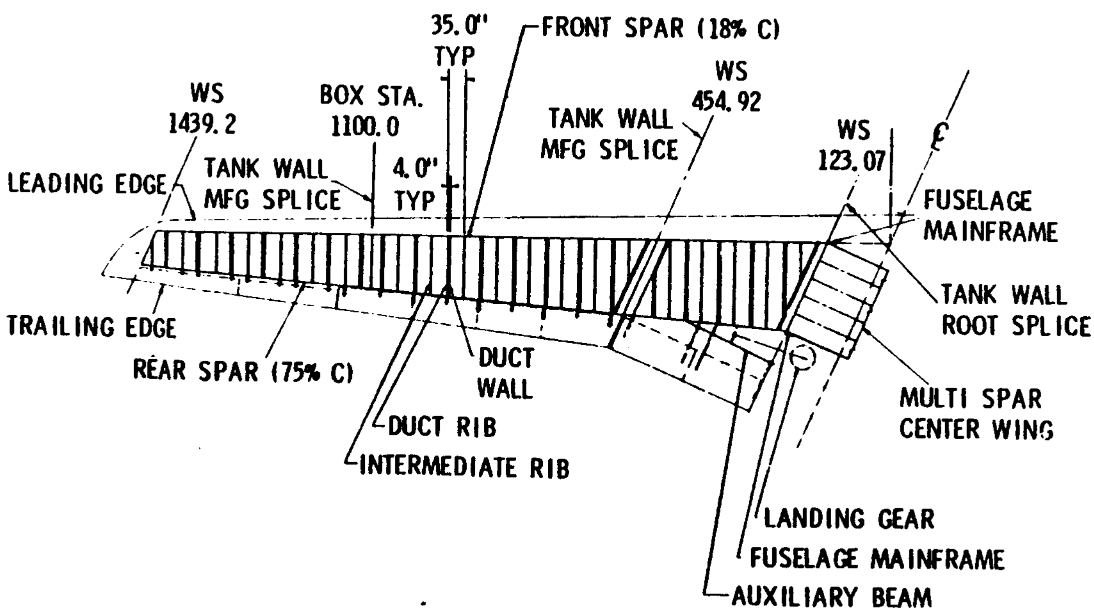

Figure 4. Wing Structural Arrangement - 1993 LFC Transport

The 18 percent chord location of the front spar was selected to provide sufficient space in the leading edge cavity for the LFC trunk ducting. The 75 percent chord location of the rear spar was selected for wing torsional stiffness reasons and to facilitate laminarization of the wing surfaces to 75 percent chord.

The leading-edge structure and ducting arrangement is shown in Figure 5. The leading edge is segmented into four sections approximately 30 ft in length. The cross section shows the two trunk lines one on each side of the trunk duct divider. The forward trunk transports the air from the upper surface and the aft trunk transports air from the lower surface.

4.1.3.1 Leading Edge LFC Slot Location

Layouts of the leading edge slots were made on a flat pattern to establish a baseline arrangement for this project as shown in Figure 6. A compromise was worked out to segment the leading edge, minimize LFC losses, and maintain a reasonable slot spacing. This configuration is shown with the chord scale 25 times the span scale.

Details of the leading edge LFC system and structure are illustrated in Figure 7. The cross sections shown in this figure are details taken from the previous figure and show sections through the spanwise duct, chordwise duct, leading edge joint and front spar attachment.

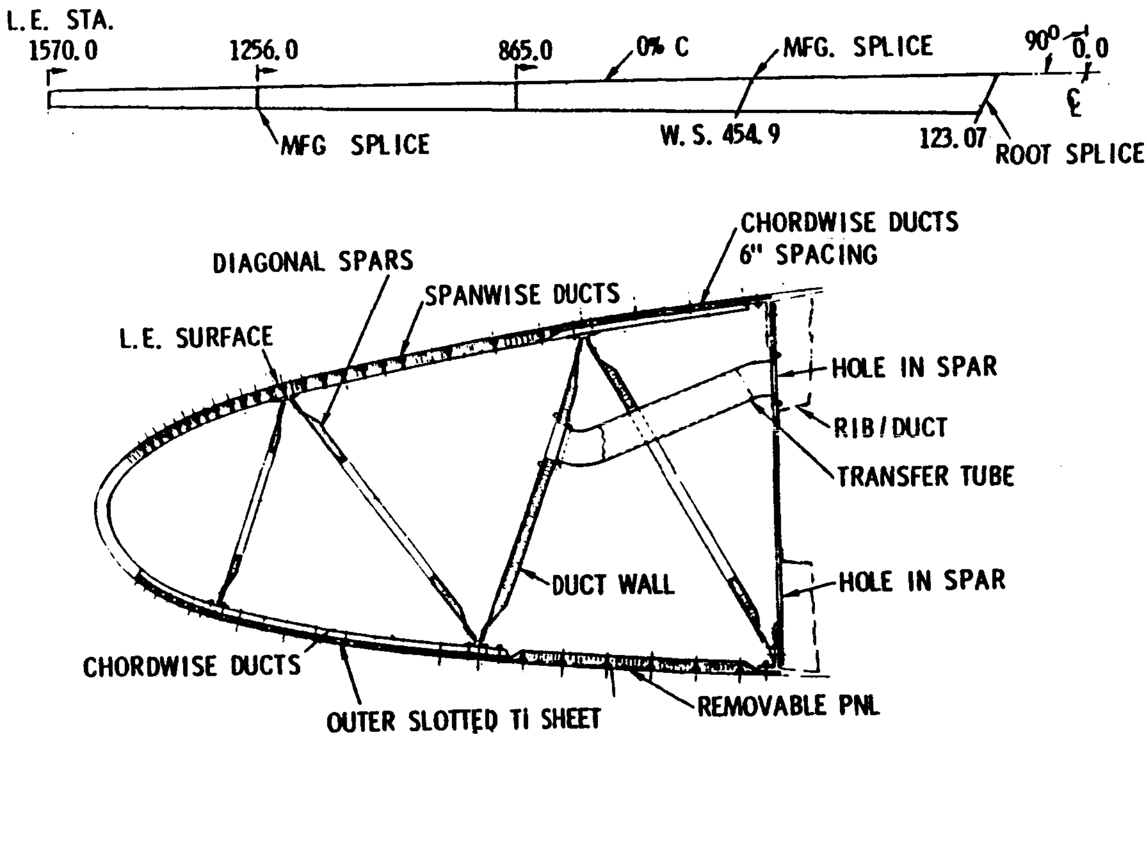

Figure 5. Leading Edge Structure & Ducting Arrangement

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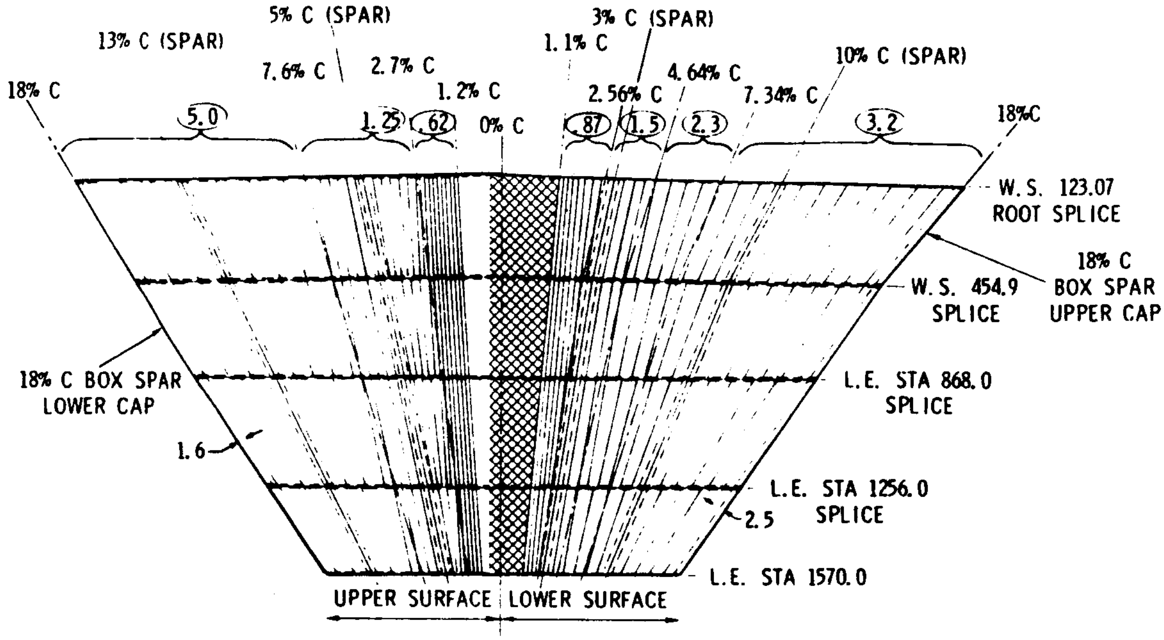

Figure 6. Leading Edge LFC Slot Locations - Flat Pattern Layout

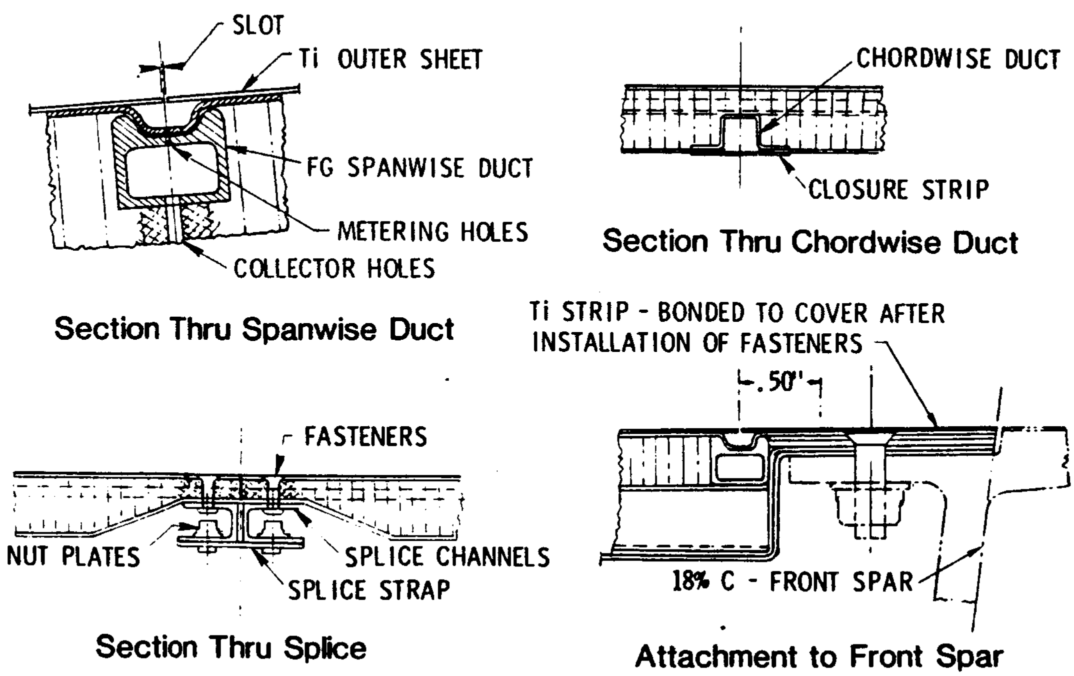

Figure 7. Leading Edge Structure - Details

4.1.3.2 Trailing Edge Flap

Figure 8 shows the preliminary arrangement of the trailing edge flaps for the 1993 LFC aircraft. There are 6 flap segments on each side of the aircraft centerline. The flap chord is 24 percent of the wing chord. The four outer segments are simple hinge type with the hinge points located aft of the rear spar and below the lower surface. The hinge brackets are cantilevered off the rear spar and each bracket supports the inboard and outboard end of adjacent flap segments. The flaps are deployed, in a direction normal to the rear spar, by screw jacks and link mechanisms which permit a differential in the angle of travel of adjacent flap segments. The flap hinges and brackets are covered by a split fairing aligned streamwise.

Each flap segment incorporates a secondary flap, with 10 percent wing chord, operated independently of the main flap segment. The secondary flap can be deployed rapidly to give additional fine tuning capacity for improved laminar flow. Each secondary flap is supported by internal hinges and is activated by an electro-hydraulic actuator.

The two inner flap segments, which also incorporate 10 percent chord secondary flaps, are deployed streamwise. They are supported by curved tracks contained within the wing contour envelope and are activated by screwjacks.

The spoilers are located on the upper surface with one segment for each flap. They are supported by hinge cantilevered from the rear spar and are hydraulically actuated.

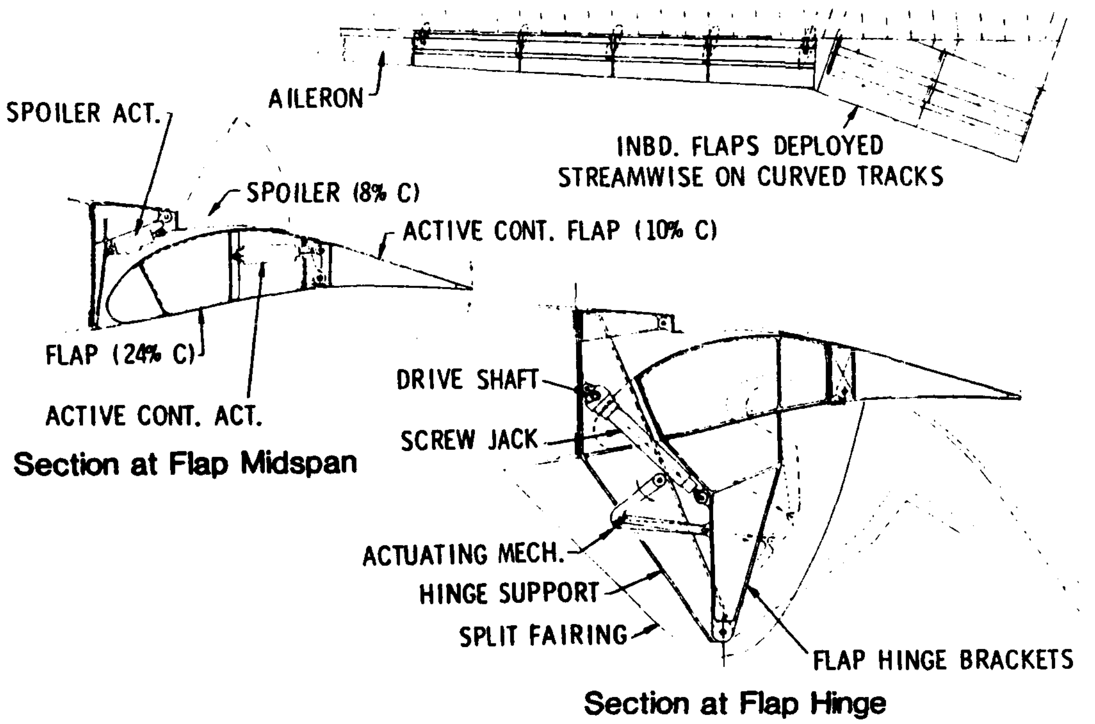

Figure 8. Trailing Edge Flap Arrangement - 1993 LFC Transport

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4.1.3.3 Wing Box Cover

The LFC wing box upper cover for the 1993 LFC transport is shown in Figure 9. The wing box upper surface extends from the W.S. 123 root splice to the W.S. 1439 tip. The cover is broken into three sections for manufacture and spliced at W.S. 454 and Box Sta. 1100. Each cover segment consists of a bonded assembly of titanium outer sheet, graphte/epoxy skin, spanwise hat stiffeners, rib caps, rib ducts, spar caps, and miscellaneous clips.

Spanwise slots, spaced 6 in apart, are cut through the titanium sheet. Slot ducts molded in the graphite/epoxy skin are located under each slot. Metering holes are drilled through the slot ducts to collect the air in the hat stiffeners.

The lower surface of the wing box is similar to the upper surface as shown in Figure 10. Slots on the lower surface are spaced 12 in apart and air is collected in each alternate spanwise hat stiffener.

Access holes are located in the lower surface, and a continuous suction slot is carried across each access door. Continuous suction slots are provided across the splices in both the upper and lower surface covers.

The surface slots and slot ducts are centered over spanwise hat-section stiffeners. The geometry of this configuration is depicted in Figure 11. Details of the manufacturing procedures planned for the hat stiffened wing box cover are discussed in a subsequent section.

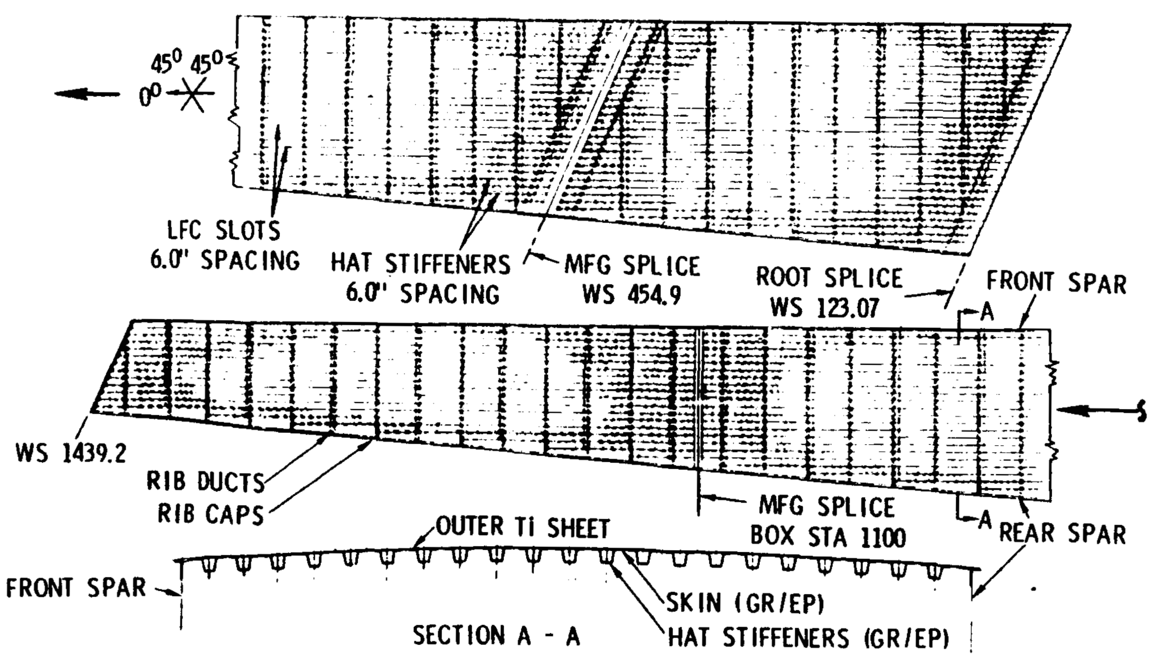

Figure 9. Wing Box Upper Cover - 1993 LFC Transport

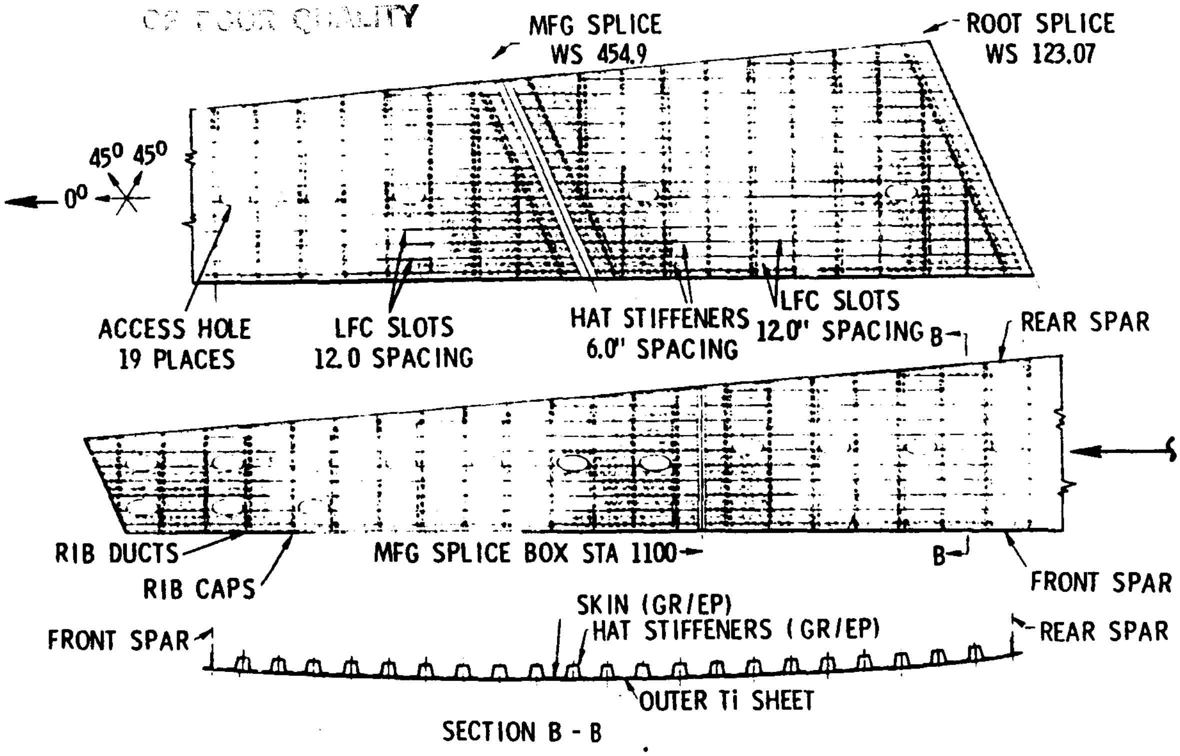

Figure 10. Wing Box Lower Cover - 1993 LFC Transport

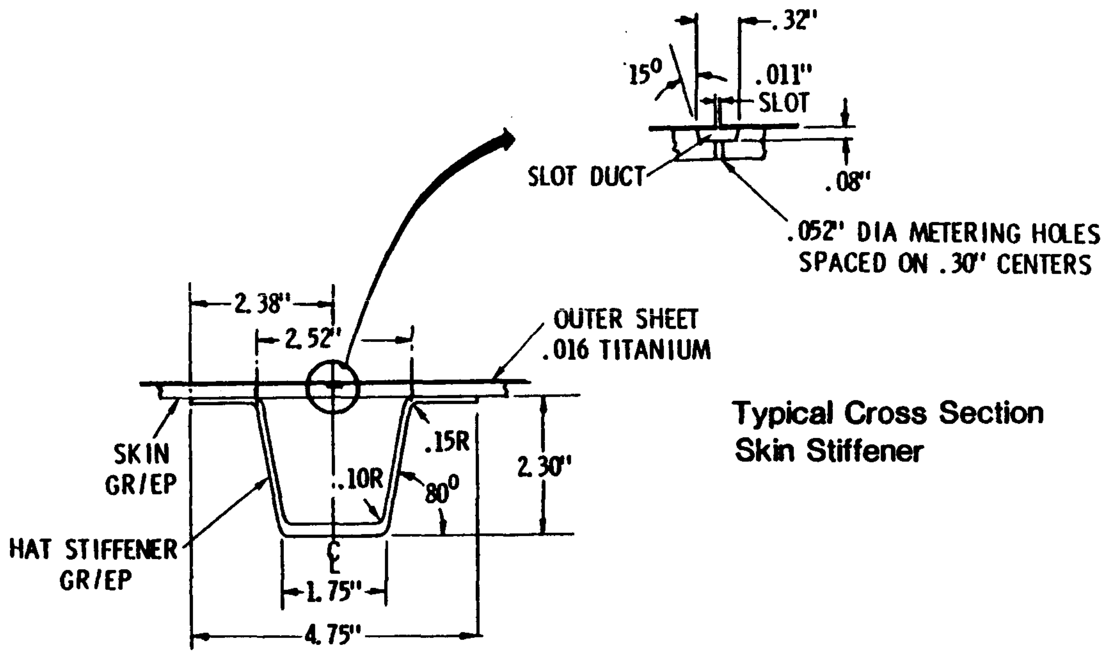

Figure 11. Typical Hat Stiffener Wing Box Cover

Details of the hat stiffened cover shown in Figure 11 are tabulated in Table I. The element thicknesses, number of plies, and ply orientations are shown for selected locations in both the upper and lower covers.

A fuel pressure of 8.0 psi (ultimate) was included in the total loading applied to the wing panels. These panels were sized to be non-buckled. Analyses of the wing covers and spars were conducted using a number of Lockheed-Georgia Company computer programs for analyzing composite structure.

Since the torsional stiffeness (GJ) requirements must be met, the approach used is to maintain the same GJ of each element of the wing box. For the wing skin, the number of $\pm 45^{\circ}$ plies was calculated by equating the GJ graphite/epoxy and GJ aluminum from the loads program. The number of 0° plies was determined by adding 0° plies until a positive margin-of-safety was shown. Skin thickness excluding the titanium are shown on Figure 12. The hat stiffener crown thicknesses are shown in Figure 13. Averge thickness, T-Bar, (including titanium skin) is plotted on Figure 14. The average panel stress (Figures 15 and 16) was calculated by dividing the panel loading by the average thickness (T-Bar). The average limit panel strains shown in Figures 17 and 18 were calculated by dividing the average stress by average moduli.

	PERCENT SEMISPAN (REF)	WING BOX STATION	SKIN			STIFF LEG			STIFF CROWN		
			THICK (IN.)	N° OF PLIES		THICK (IN.)	N° OF PLIES		THICK (IN.)	N° OF PLIES	
				0°	45°		0°	45°		0°	45°
UPPER COVER	14.5	274	.245	25	24	.09	2	16	.25	42	8
	25.3	445	.305	23	38	.09	2	16	.24	40	8
	35.3	607	.315	19	44	.09	2	16	.24	40	8
	45	762	.275	15	40	.09	2	16	.18	28	8
	55	918	.220	10	34	.09	2	16	.18	28	8
	65	1080	.175	9	26	.07	2	12	.12	16	8
	75	1238	.115	5	18	.065	1	12	.10	12	8
	85	1396	.065	5	8	.065	1	12	.06	8	4
	95	1555	.050	5	6	.065	1	12	.06	8	4
LOWER COVER	14.5	274	.215	15	28	.09	2	16	.25	42	8
	25.3	445	.265	13	40	.09	2	16	.24	40	8
	35.3	607	.305	9	52	.09	2	16	.24	40	8
	45	762	.245	7	42	.09	2	16	.18	28	8
	55	918	.195	3	36	.09	2	16	.18	28	8
	65	1080	.155	3	28	.07	2	12	.12	16	8
	75	1238	.115	3	20	.065	1	12	.10	12	8
	85	1396	.075	3	12	.065	1	12	.05	6	4
	95	1555	.045	3	6	.065	1	12	.02	2	2

TABLE 1.
COVER ELEMENT THICKNESSES AND NUMBER OF PLIES

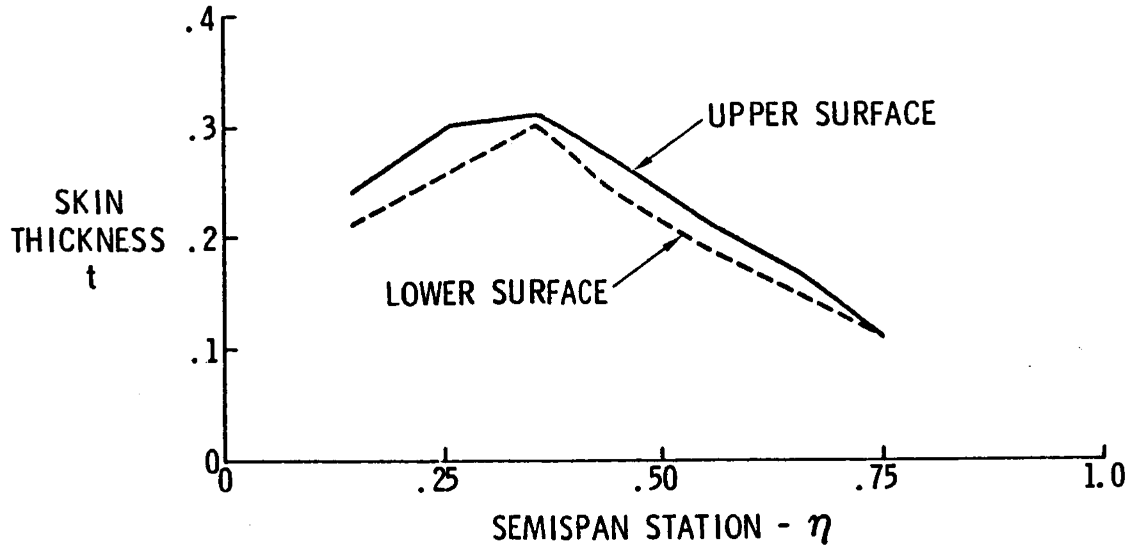

Figure 12. LFC Wing Panel Thickness

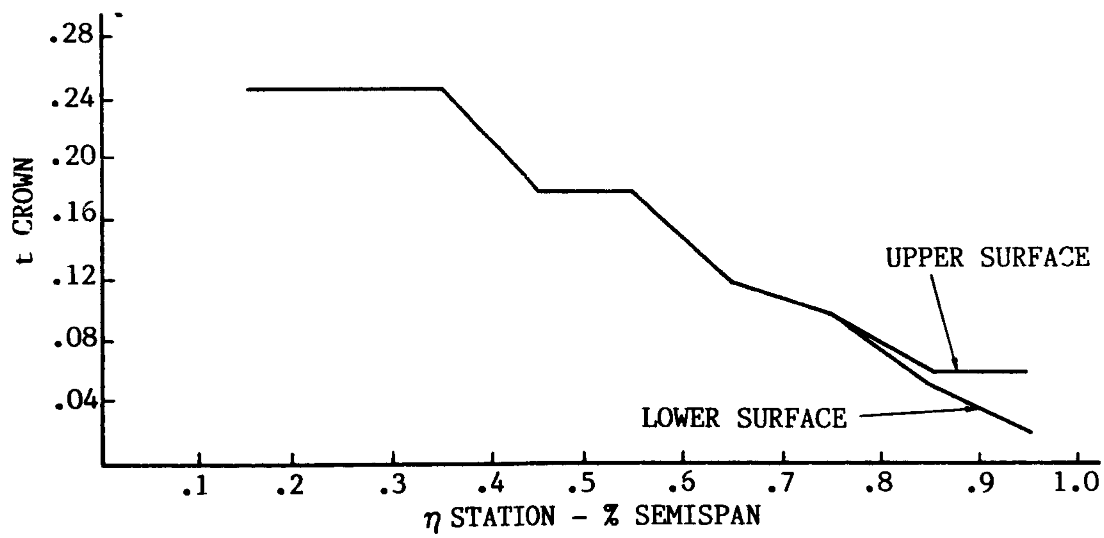

Figure 13. LFC Wing Panel Stiffener Crown Thickness

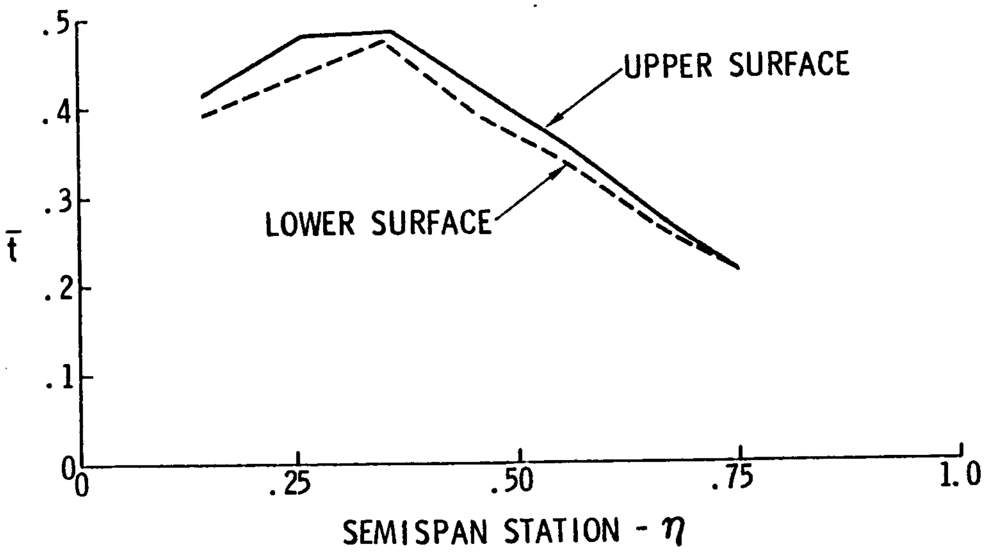

Figure 14. Average Thickness of Wing Panel - $\bar{t}$

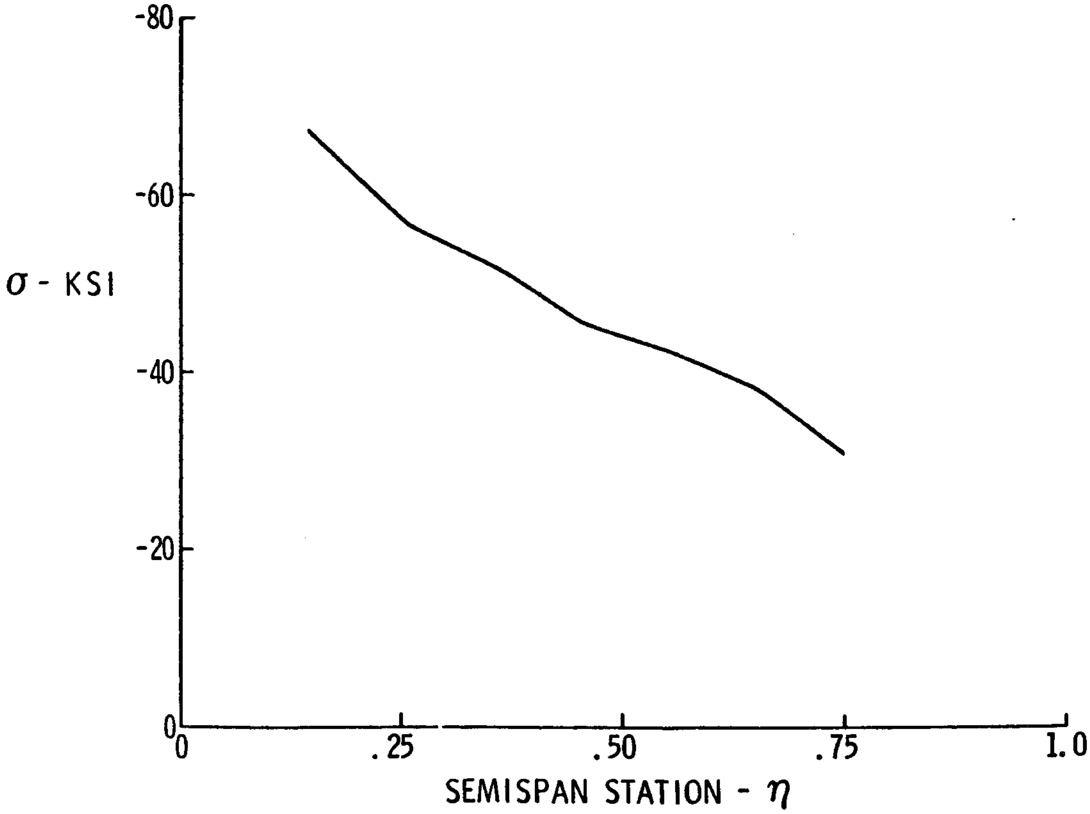

Figure 15. Average Stress in Upper Surface of Wing Panel

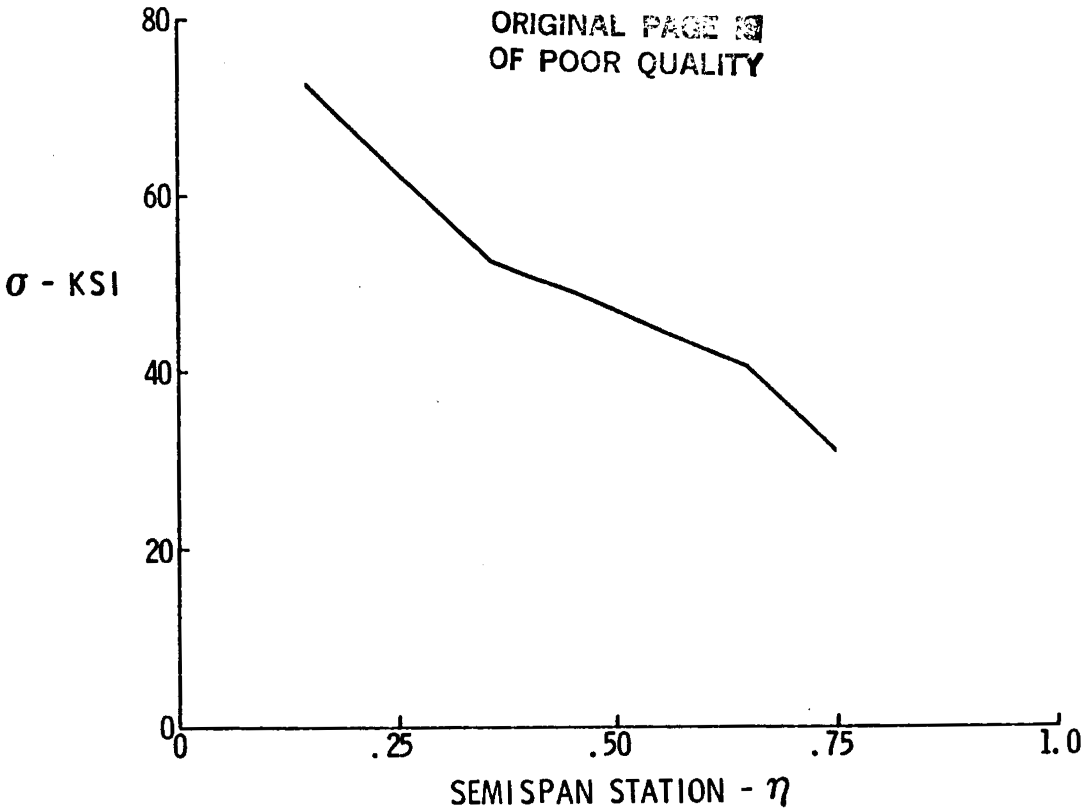

Figure 16. Average Stress in Lower Surface of Wing Panel

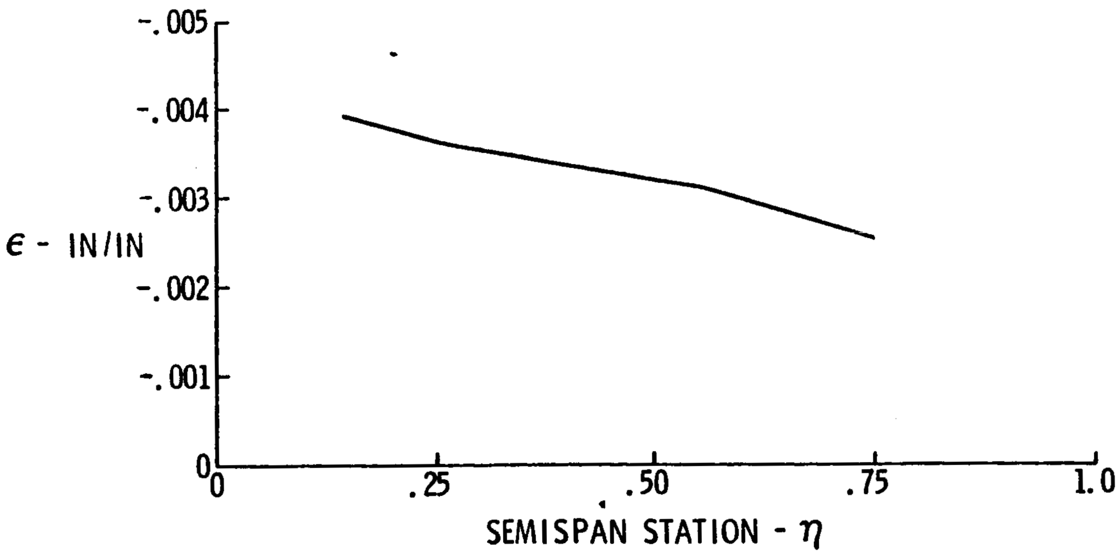

Figure 17. Limit Strain in Upper Surface of Wing Panel

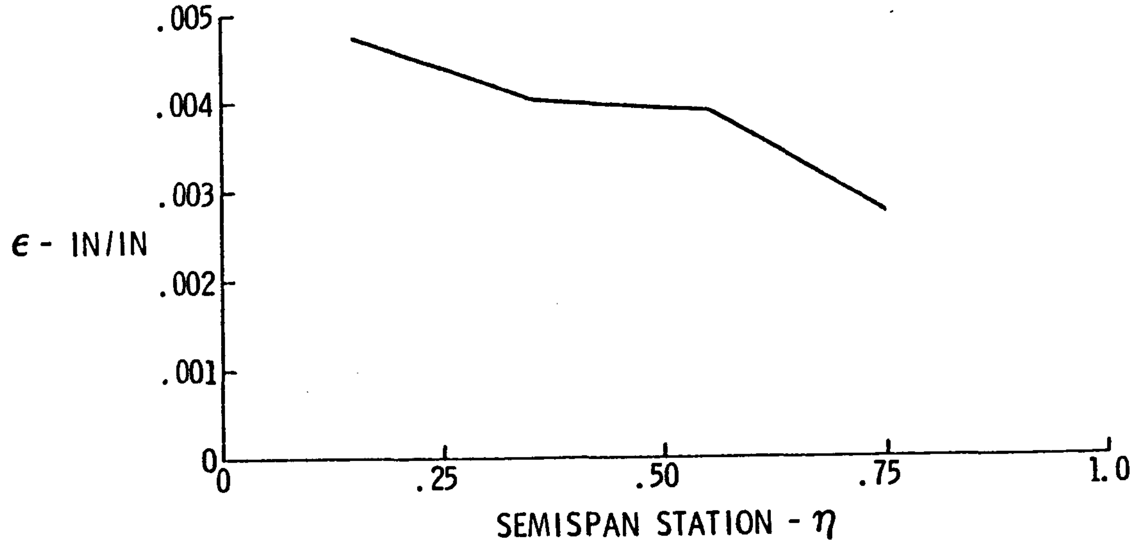

Figure 18. Limit Strain in Lower Surface of Wing Panel

4.1.3.4 Typical Duct Rib Wing Box Structure

Figure 19 shows the general configuration of a typical, duct-type rib and details of its assembly. The rib is comprised of upper and lower box-cross-section caps, which are supported by a system of vertical posts and diagonal braces mechanically attached to the caps. All members are fabricated from graphite/epoxy material. The rib caps and bracing members are sized by compressive loads in the surface panels. The caps provide support for the panels and the bracing members support the caps. The maximum compressive load in the upper surface panel is of greater magnitude than for the lower thus the upper rib cap requires a larger number of supports than the lower cap. The post and diagonal brace arrangement shown in Figure 19 satisfy this requirement.

A typical cross section through the rib cap and the chordwise duct for the LFC system is shown in Figure 19. The cap is comprised of the "Tee" cross-section, primary cap member, which reacts all the bending loads and a "Zee" cross-section duct wall located 4 in from the primary cap. These members are bonded to the surface panel skins and stiffeners and are an integral part of the surface panel assembly. The box section is completed by an upper closure member which is mechanically attached to the primary cap and to the duct wall after the bracing members are installed. This closure can be removed for inspection or repair if required. Blind fasteners are not required for this assembly.

Figure 20 shows typical details of the duct rib. Details of the bracing members, rib cap, molded clips and attachments are shown. The forward and aft cuts of the rib ducts are attached to the front and rear spars and sealed from the fuel contained in the wing box structure.

The arrangement of the intermediate ribs is similar to the arrangement of the duct ribs except the duct wall is not required, and the closure member is replaced by a single angle bonded to the primary cap member.

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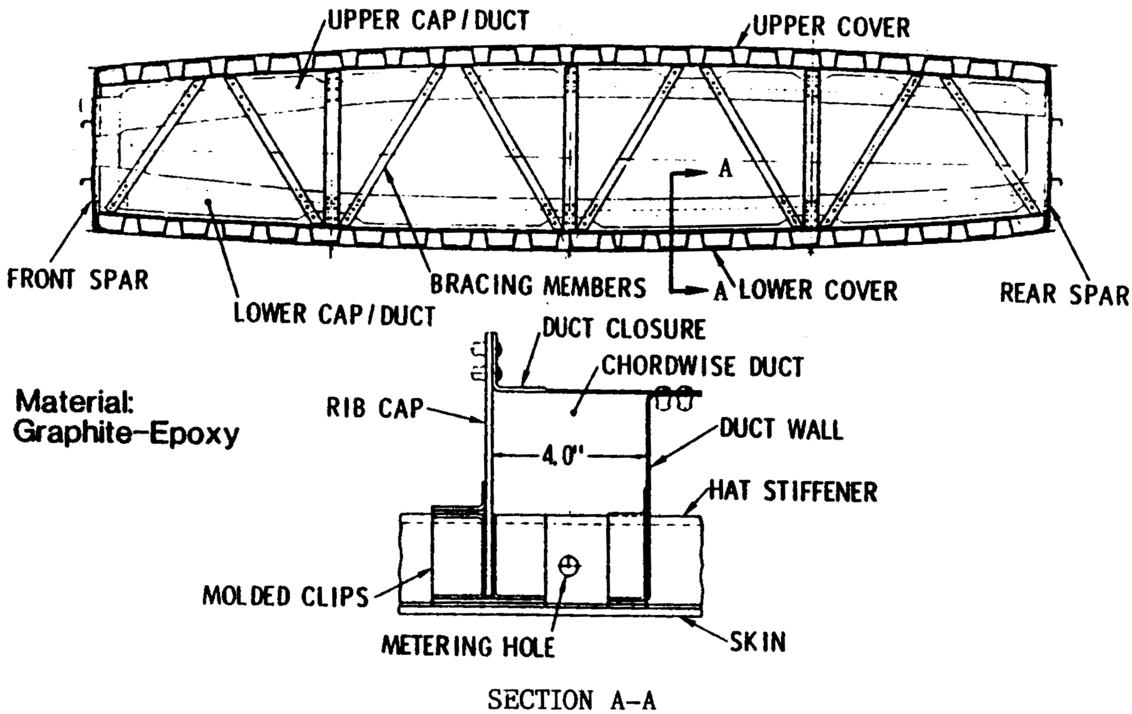

Figure 19. Typical Duct Rib Wing Box Structure

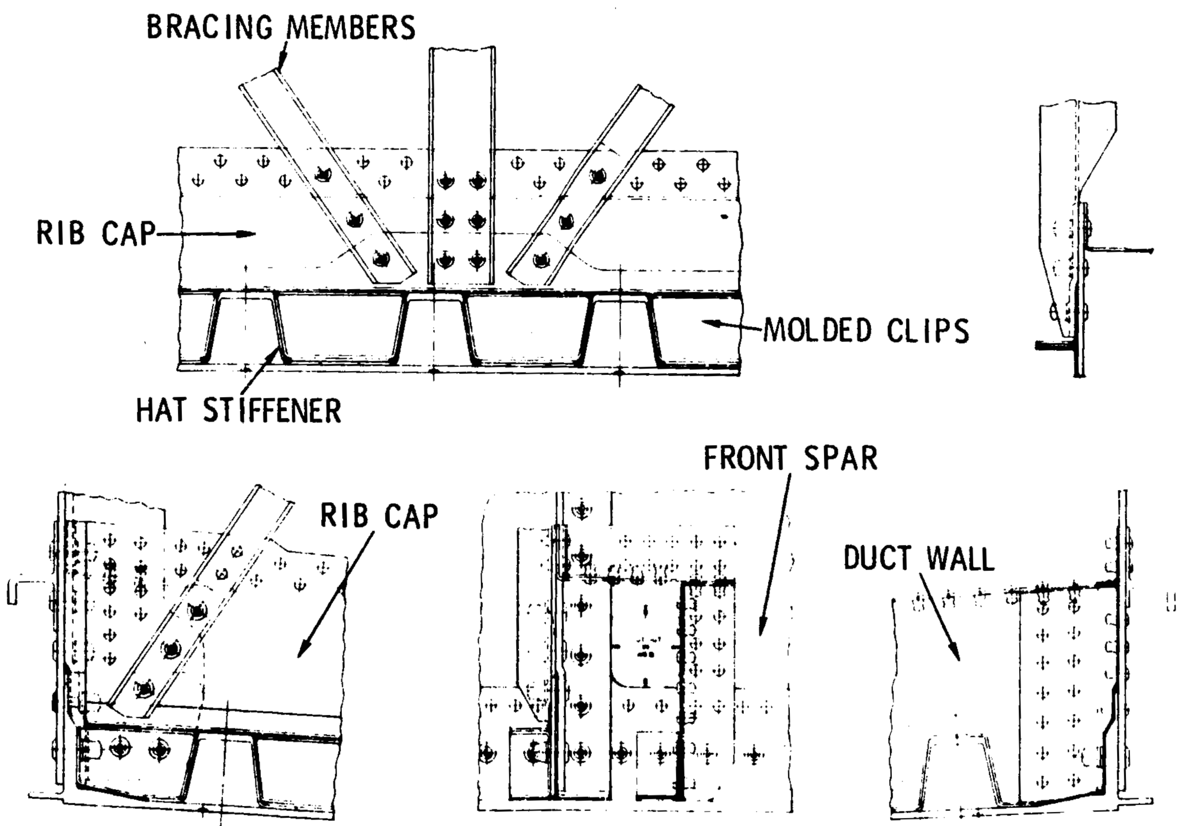

Figure 20. Typical Details of Duct Rib

4.1.3.5 Wing Box Structural Spars

The general arrangement and details of the wing-box front spar are shown in Figure 21. Studies were conducted to determine the arrangement of the spar web stiffeners and the web location relative to the spar cap vertical leg. Both vertical and spanwise stiffener arrangements were considered. The study results indicated that the best arrangement was with the web located on the forward side of the spar cap vertical leg and with the stiffeners aligned spanwise as shown in Figure 21. The spanwise stiffener arrangement at the front spar offers the least disturbance to LFC air flowing through the leading-edge cavity.

The front spar in the Figure 21 depicts the numerous metering holes required to transfer LFC air from the box structure chordwise ducts to the leading edge trunk ducts. These holes vary in diameter from 3.5 in at the root to 1.9 in at the tip for the lower surface and from 2.30 in to 1.25 in for the upper surface.

The general arrangement of the rear spar of the wing box structure is shown in Figure 22. The figure shows the location of the flap hinge brackets and the large fittings required to support the landing gear trunion and the outboard end of the auxiliary spar. The spar web thicknesses shown in Figure 23 and Figure 24 were based on two assumptions:

- o The spar web GJ would be maintained.
- o 90 percent of the spar web would be $\pm 45^{\circ}$ plies and 10 percent would be 0° plies .

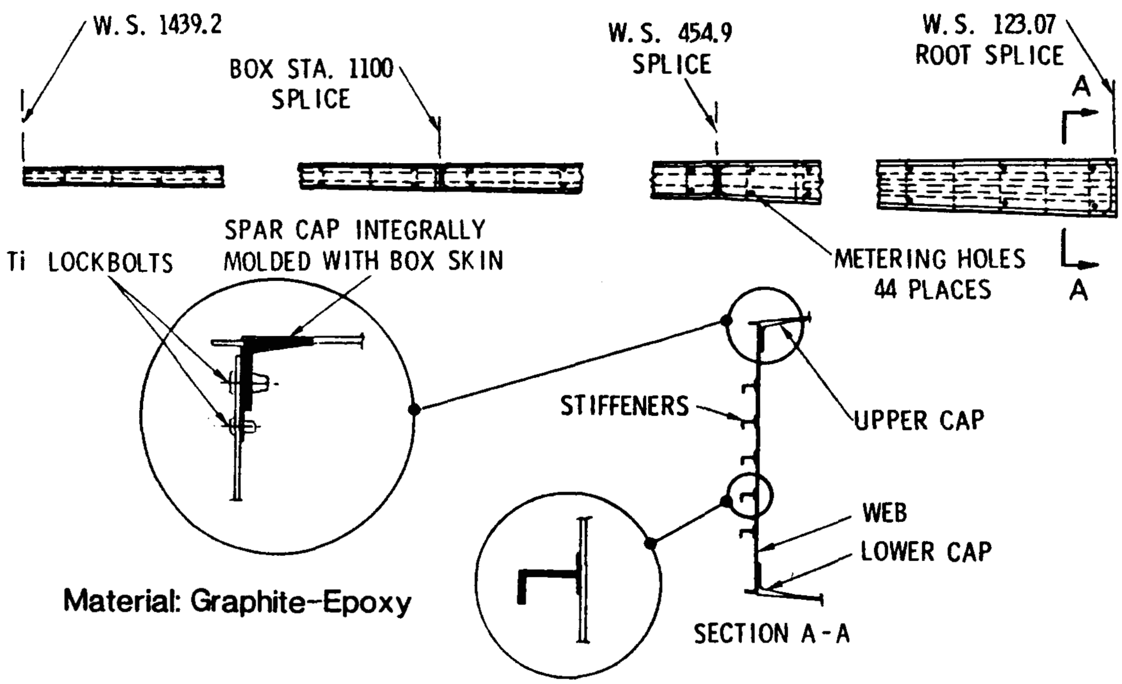

Figure 21. Front Spar Wing Box Structure

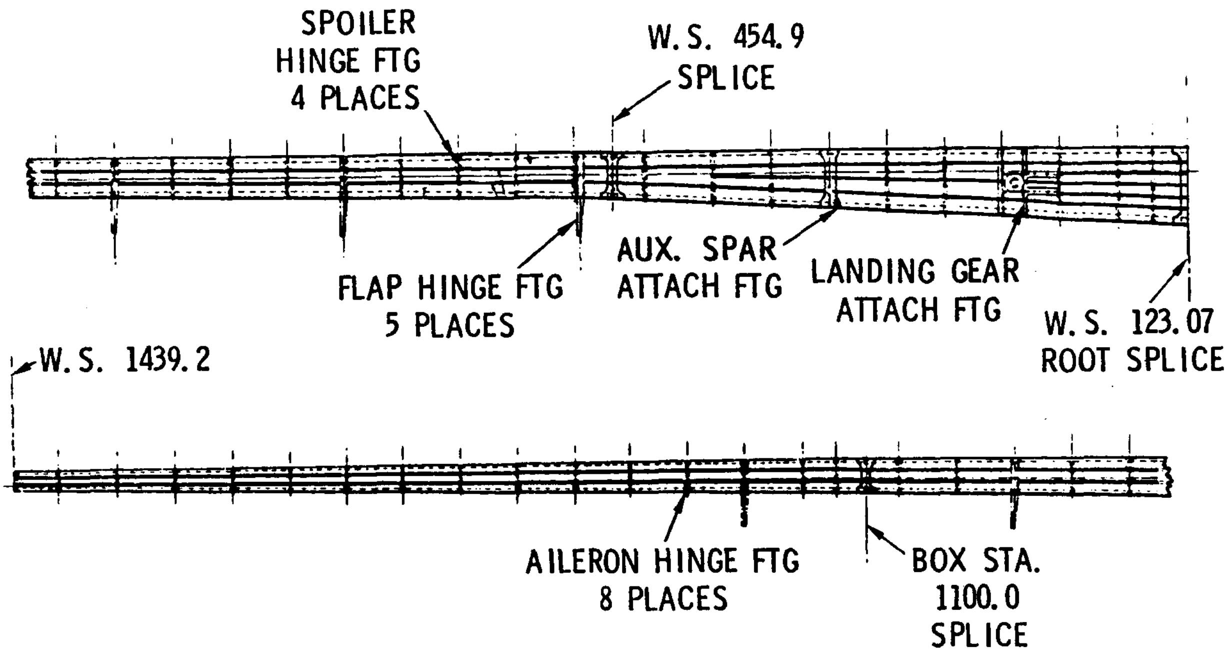

Figure 22. Rear Spar Wing Box Structure

	FRONT SPAR			REAR SPAR		
SEMI-SPAN	T ±45° PLIES	T 0° PLIES	T_{TOTAL}	T ±45° PLIES	T 0° PLIES	T_{TOTAL}
.95	.042	.005	.046	.042	.005	.046
.85	.100	.012	.112	.100	.012	.112
.75	.139	.016	.155	.139	.016	.155
.65	.183	.021	.204	.183	.021	.204
.55	.224	.026	.250	.224	.026	.250
.45	.260	.030	.290	.260	.030	.290
.353	.294	.034	.328	.294	.034	.328
.253	.256	.030	.285	.256	.030	.285
.145	.192	.022	.214	.192	.022	.214

Figure 23. Spar Webs - Thickness (In.)

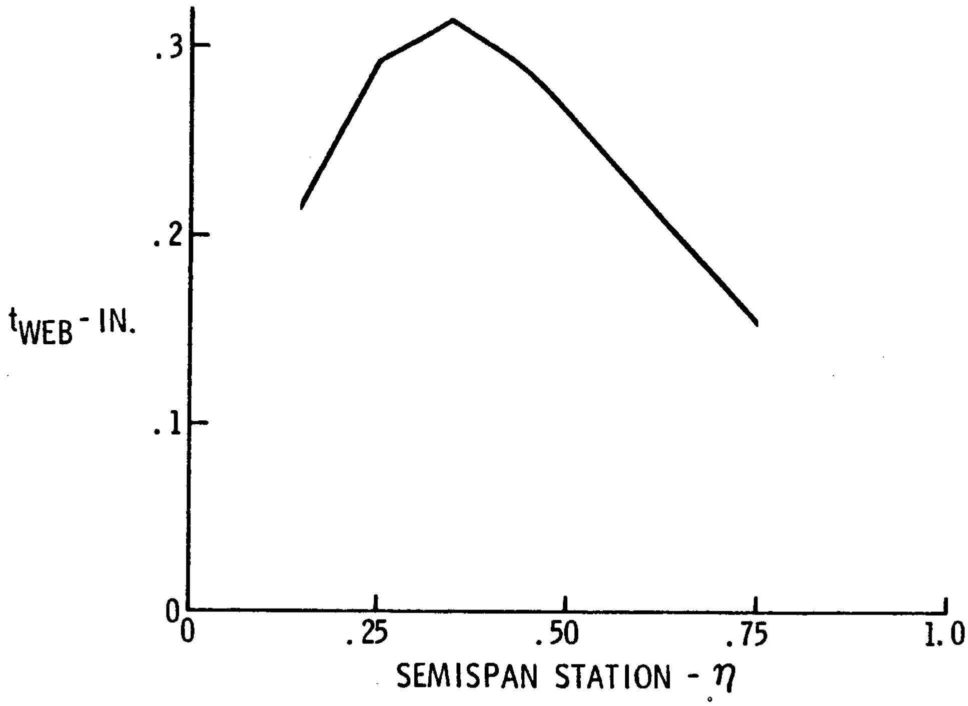

Figure 24. Thickness of Wing Spar Web

Stiffener spacing was varied such that the web did not buckle up to ultimate loads. Allowable buckling loads were calculated using computer program LG-031. The applied ultimate and allowable spar web shears are shown in Figure 25.

For the baseline wing the spar caps were assumed to be attached with mechanical fasteners. The spar caps to cover skins and to spar web fasteners were sized to carry the spar web allowable buckling shear. Two rows of fasteners were used with a spacing of 5D (5 times diameter) and an edge distance of 2.5D (2.5 times diameter). An allowable graphite/epoxy bearing stress of 80 ksi was used. Spar cap fastener diameters are shown in Figure 26.

Flange widths of the spar caps were set by the fastener spacing and edge distance requirements. The number of $\pm 45^\circ$ plies was kept the same in the spar webs as in the cover to maintain the torsional stiffness of the box. The cap allowable buckling loads were increased by increasing the number of 0° plies. For tension-critical caps, extensional stiffness was calculated and each element analyzed for its proportional share of the load based on a ratio of element stiffness divided by total section stiffness. The upper and lower spar cap leg lengths and thicknesses are shown in Figures 27 and 28, respectively.

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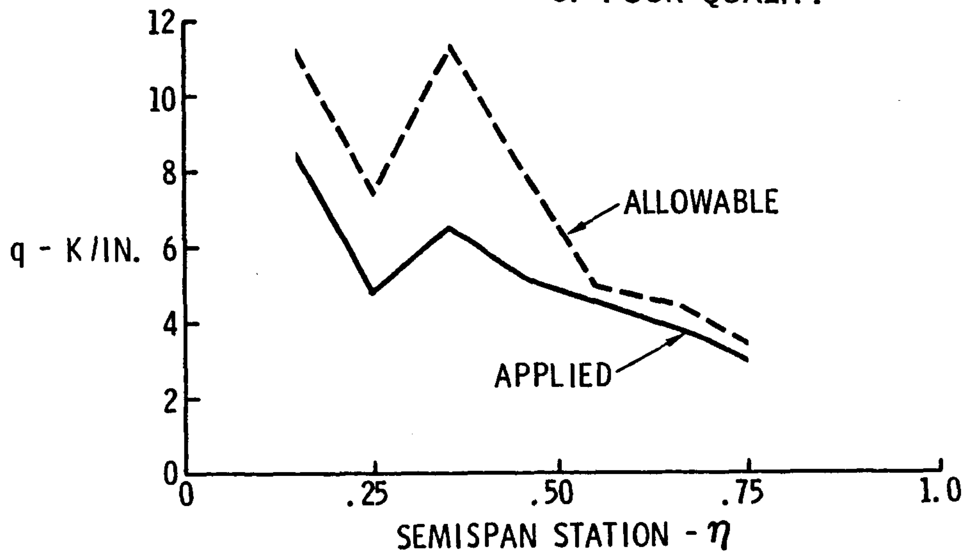

Figure 25. Applied and Allowable Shear in Spar Web

SEMI-SPAN	FRONT SPAR FASTENER DIAMETER (IN.)	REAR SPAR FASTENER DIAMETER (IN.)
.95	.156	.156
.85	.156	.156
.75	.156	.156
.65	.188	.188
.55	.188	.188
.45	.313	.313
.353	.375	.250
.253	.313	.250
.145	.375	.313

NOTE 1. 2 ROWS @ 5D SPACING W/2.5 E/D

Figure 26. Spar Cap Attachment Diameter

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SEMI-SPAN	FRONT SPAR CAP LEG LENGTH (IN.)	NO. PLIES 0°	NO. PLIES ±45°	THICKNESS TITANIUM (IN.)	THICKNESS OF CAP (IN.)	REAR SPAR CAP LEG LENGTH (IN.)	NO. PLIES 0°	NO. PLIES ±45°	THICKNESS CAP (IN.)
.95	1.65	2	16	-	.090	1.63	2	12	.070
.85	1.09	5	20	-	.125	1.67	2	20	.110
.75	1.73	6	28	-	.170	1.72	3	28	.155
.65	2.10	8	36	-	.220	2.08	4	40	.220
.55	2.19	18	44	-	.310	2.2	5	44	.245
.45	3.48	-	52	.093	.353	3.42	6	52	.290
.353	4.14	-	56	.11	.390	2.83	10	56	.330
.252	3.57	36	52	-	.440	2.87	22	52	.370
.145	4.26	42	40	.11	.520	3.63	62	40	.510

Figure 27. Upper Spar Caps Legs Length and Thickness

SEMI-SPAN	FRONT SPAR CAP LEG LENGTH (IN.)	NO. PLIES 0°	NO. PLIES ±45°	THICKNESS TITANIUM (IN.)	THICKNESS OF CAP (IN.)	REAR SPAR CAP LEG LENGTH (IN.)	NO. PLIES 0°	NO. PLIES ±45°	THICKNESS CAP (IN.)
.95	1.65	2	16	-	.090	1.63	2	12	.070
.85	1.67	2	20	-	.110	1.67	2	20	.110
.75	1.72	3	28	-	.155	1.72	3	28	.155
.65	2.08	4	36	-	.200	2.08	4	36	.200
.55	2.13	6	44	-	.250	2.13	6	44	.250
.45	3.48	-	52	.093	.353	3.42	6	52	.290
.353	4.14	-	56	.110	.390	2.81	6	56	.310
.253	3.51	24	52	-	.380	2.82	12	52	.320
.145	4.16	44	40	-	.420	3.51	38	40	.390

Figure 28. Lower Spar Caps Legs Length and Thickness

4.2 DESIGN CONCEPTS

LFC design concept studies were conducted in the following areas:

- o Wing rib/cap duct
- o Wing spar cap
- o Wing chordwise surface splice
- o Wing spar web stiffener

The conceptual designs were evaluated relative to the criteria as listed:

- Suction duct efficiency
- Weight
- Cost
- Structural integrity
- Manufacturing
- Maintainability
- Design feasibility

Appropriate evaluators in each area were asked to evaluate each concept in regard to their specialty and to score them on a basis of 0 to 10 (10 being the highest). Each evaluator was required to comment on any scores between 0 and 3 and between 7 and 10. The purpose of this requirement was twofold:

(1) For the low scores, to ensure that a simple redesign would not eliminate the problem areas.

(2) For the high scores, to investigate the possibility of incorporating these into other designs.

The evaluators were also requested to place the concepts in preferential rank.

To aid them in these evaluations, each evaluator was supplied with a data package containing illustrations and a written description of each concept, and also a listing of possible problem areas which should be considered in the evaluations. A typical evaluation procedure for concept selection is shown by the blank table in Figure 29.

4.2.1 Wing Rib/Cap Duct

The chordwise ducts transfer air from the spanwise hat stiffeners to the main collector ducts in the leading edge of the wing. Suction ducts are located at each alternate rib location along the wing span at both the upper and lower wing surfaces. The baseline LFC wing has a total of 44 chordwise ducts. Duct lengths range from 204 in at the wing root to 80 in at the wing tip.

The height of the rib duct tapers from approximately 8 in at the front spar near the wing root to approximately 0.5 in at the rear spar. The width of the ducts is approximately 4 in. These dimensions are reduced in the outboard section of the wing.

Twenty-nine alternative duct concepts were investigated during the study. The objective of the study was to select the configuration which best satisfies the requirements for duct suction efficiency, minimum cost and weight,

EVALUATION CRITERIA	(SCORE 0 - 10) (RANK 1 - 5)									
	1		2		3		4		5	
	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE
1. SUCTION DUCT EFFICIENCY										
2. WEIGHT										
3. COST										
4. INTEGRITY										
5. MANUFACTURING										
6. MAINTAINABILITY										
7. DESIGN FEASIBILITY										
TOTAL SCORE										
RANK										

Figure 29. Typical Evaluation Procedure for Concept Selection

structural integrity, producibility, maintainability, inspectability, and design feasibility. The concepts are grouped into three categories. Group 1: concepts 1 through 15 shown in Figure 30 have twin rib caps spaced 4 in apart. Group 2: concepts 16 through 25 shown in Figure 31 have a single rib cap with an adjacent chordwise duct. Group 3: concepts 26 through 29 shown in Figure 32 have the chordwise duct completely separate from the rib.

During a preliminary screening of the configurations in Figures 30 and 31 fifteen were eliminated because of readily evident deficiencies. Concepts 1, 2, 3, 8, 12 and 13 were eliminated because access into duct cavity, for inspection, maintenance and repair, could not be achieved without disassembling the rib. Concepts 5, 6, 10, 21 and 22 were eliminated because of the large number of blind fasteners which would be required. Concepts 26, 27, 28 and 29 were eliminated because these require more parts than other concepts. Hence, these would be heavier and costlier. Also, these locations were somewhat constrained by the need for access panels in the lower cover.

The remaining configurations were subjected to a further screening process.

The summary of the final rank/score are shown in Figure 33 for the fourteen concepts. Concept No. 16 received the final rank of "number one" for both rank and score and was the chosen concept.

4.2.2 Wing Spar Cap

Six different spar cap concepts were evaluated. These are illustrated in Figures 34 and 35 described below:

~~Concept 1~~ - The spar caps are integrally molded with the spar web and attached to the box covers with mechanical fasteners. Titanium shims are laminated in the caps and box covers to increase bearing strength in highly loaded areas.

Concept 2 - Separate spar caps extensively machined from titanium extrusions are used. They are attached to the covers and to the spar web with mechanical fasteners. Some titanium shim embedments are required in the box covers, and in the spar web, to increase bearing strength for the fasteners in certain highly loaded areas.

Concept 3 - The spar caps are separate, graphite-epoxy, molded parts. The caps are bonded to and mechanically attached to the box covers, and mechanically attached to the spar webs. Titanium shim embedments are provided, in both horizontal and vertical spar cap legs, to increase bearing strength in certain highly loaded areas. The box covers and spar web are also similarly reinforced for the same reasons.

Concept 4 - Spar caps integrally molded with the graphite-epoxy box covers are used. The spar web is attached to the vertical leg of spar cap with mechanical fasteners. Fasteners through the horizontal cap and cover are eliminated, thus titanium shim embedments are required only in the spar cap vertical leg and spar web to increase bearing strength in certain highly loaded areas.

Concepts 5 and 6 - These concepts are variations of Concept 4 and are essentially the same except for the cross sectional shapes of the cap. These shapes were derived to eliminate joggling of the stiffener flange at a transition point located approximately 50 percent semispan.

In evaluating these concepts, due consideration was given to the interface of the spar caps with adjacent structure such as box covers, rib cap ducts, rib attachment members, and the chordwise splices. The type of cap selected may affect the method of assembly and the fabrication of the covers.

<u>Concept 1: Assembly Sequence</u> - It is desirable that the stiffeners and rib caps be bonded in place in the box cover assemblies. A slot must be provided in the forward edge of each cover to permit insertion of the spar caps. Although this concept is attractive from a spar fabrication viewpoint it will be difficult to assemble due to tolerances of spar cap thickness and slot dimension, and will also be more difficult to seal. A further disadvantage is that the titanium outer sheet cannot be installed, or the LFC slots sawed until the cap-cover fasteners are installed at the final assembly stage.

<u>Concepts 2 and 3 Assembly Sequence</u> - The spar cap can be bonded to the covers together with the stiffeners and rib cap/ducts. Cap to cover fasteners are then installed, the outer titanium sheet bonded in place, and the LFC slots sawed to complete the box cover assembly. The upper and lower covers are then located in a jig and rib truss members and rib attachment members are installed. The spar web, located on the forward faces of the caps, is then installed to complete the wing box assembly. This location of the web ensures easy removal for replacement in case of damage, and also provides better access into the box for assembly of the members.

Concepts 4, 5 and 6 Assembly Sequence - The assembly sequence is similar to Concepts 2 and 3 and all of the advantages and disadvantages also apply. In these concepts, however, the spar caps are integrally molded with the box covers, thus eliminating the need for mechanical fasteners and titanium shims in the horizontal leg of the caps.

The caps and covers are spliced at chordwise joints located at the root (Box Sta 134) at the bat break (Box Sta 496) and at Box Sta 1100. Splices are accomplished using bonding and mechanical fasteners. The covers will be double shear spliced with chordwise plates located at the inner and outer surface of each cover. Each cap will be spliced by means of a metallic fitting located at the inner surface of the cap. Localized titanium inserts will be required in the composite parts due to fastener bearing considerations. The type of spar cap selected may influence the complexity of the splice. For example, in Concepts 1 and 3, titanium shim inserts would be required in both cover and cap. In Concepts 4, 5, and 6, only a single insert may be required. The selection of Concept 5 would require a more complex stepped splice fitting because of lack of space between the spar cap and the hat leg.

Figure 36 shows the results of the spar cap evaluation. Concept number 6 received the final rank of "number one" for both rank and score and was chosen for the baseline concept.

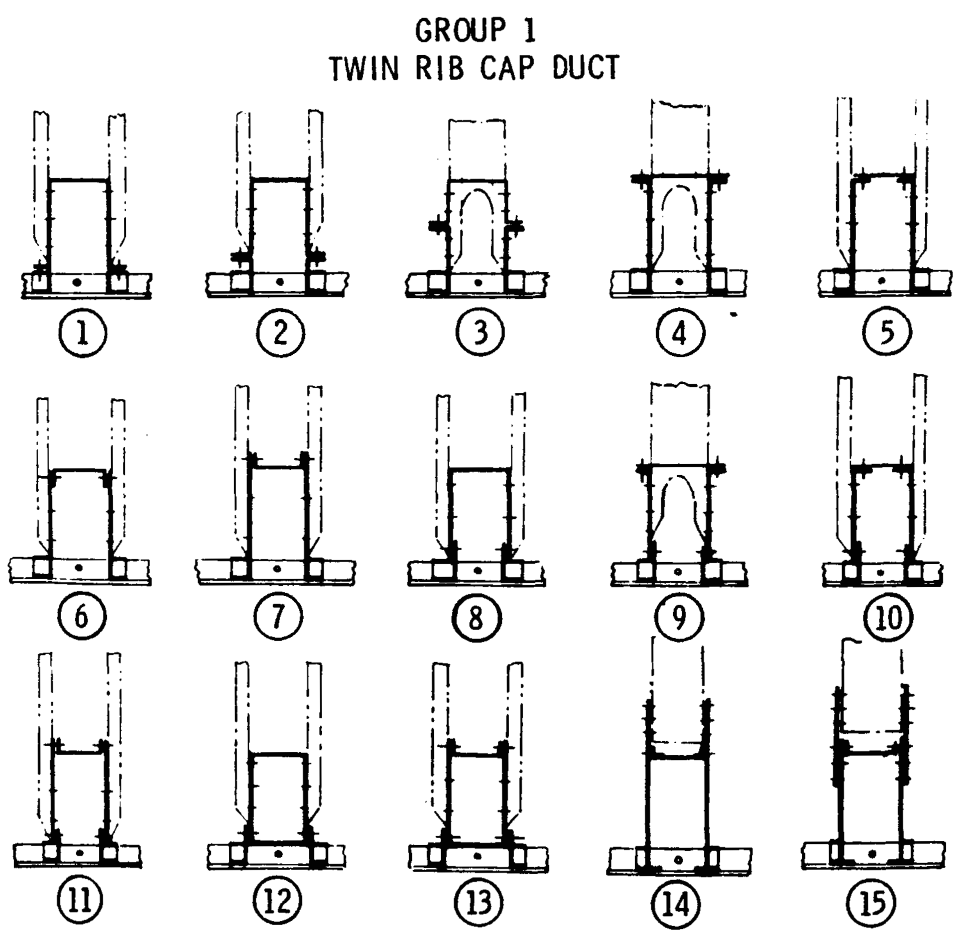


Figure 30. Rib Cap/Chordwise Duct Concept Alternatives

GROUP 2
SINGLE RIB CAP - ADJACENT DUCT

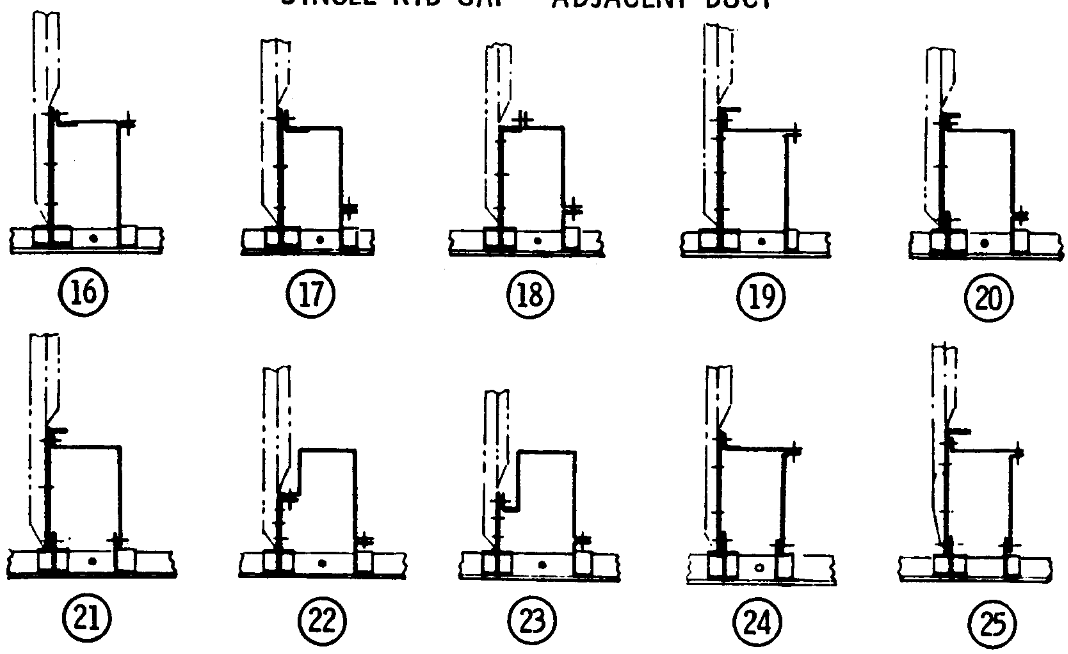


Figure 31. Rib Cap/Chordwise Duct Concept Alternatives
Group 2

GROUP 3
INDEPENDENT DUCT

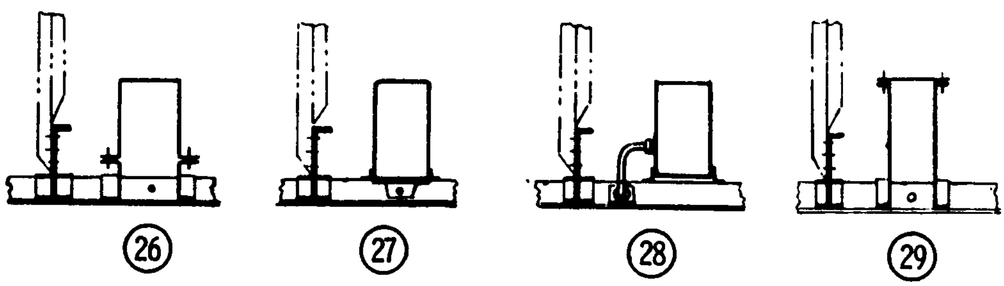
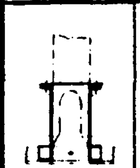
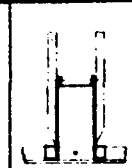
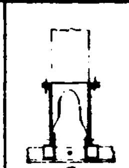
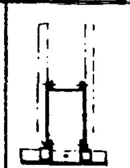
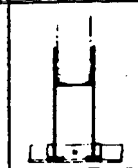
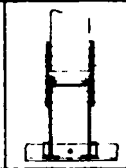
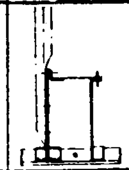


Figure 32. Rib Cap/Chordwise Duct Concept Alternatives
Group 3

CONCEPT NO.	4		7		9		11		14		15		16	
														
RANK / SCORE	R	S	R	S	R	S	R	S	R	S	R	S	R	S
TOTAL	55	29	36	42	67	25	54	33	42	38	56	32	34	45
FINAL RANK	8	11	2	4	12	14	7	9	5	5	9	10	1	1

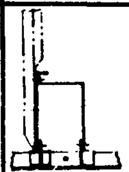
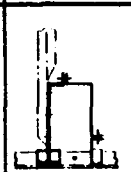
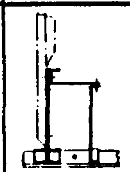
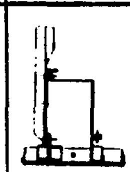
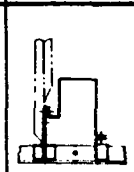
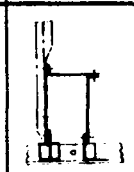
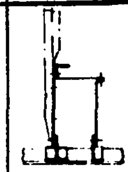
CONCEPT NO.	17		18		19		20		23		24		25	
														
RANK / SCORE	R	S	R	S	R	S	R	S	R	S	R	S	R	S
TOTAL	34	44	49	38	37	45	56	35	63	34	69	27	67	28
FINAL RANK	4	3	6	5	3	1	9	7	11	8	14	12	12	12

Figure 33. Rib Cap/Duct Concept Selection

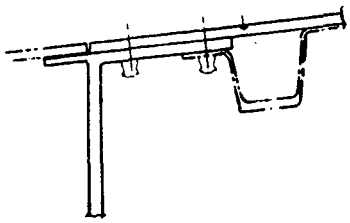
4.2.3 Wing Chordwise Splice

The upper and lower box covers for both the left and right wings are spliced at:

- o Side of Body - WS 123 (Surface Crease)
- o Outer wing/bat - WS 455 (Surface Crease)
- o Outer wing - BS 1100 (Production Joint)

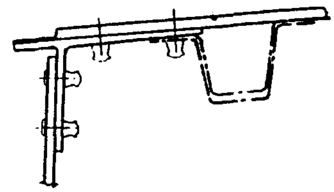
The box covers are provided with spanwise slots for the LFC system and, to achieve maximum LFC efficiency, the slots must be continuous across the splices at Wing Station 455 and at Box Station 1100. At the wing root, however, the LFC slots are terminated prior to the splice hence the splice plates do not require slots at this location.

Seven concepts were subjected to a screening process in which a number of technical specialists were asked to evaluate, score, and rank each concept for suction duct efficiency, weight, cost, producibility, maintainability and structural integrity. A data package, containing illustrations, a written description of each concept, and a listing of possible problem areas which should be considered, were given to each evaluator.



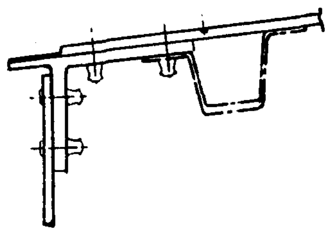
CONCEPT 1

SPAR CAP INTEGRALLY
MOLDED WITH SPAR WEB



CONCEPT 2

SEPARATE TITANIUM SPAR CAP



CONCEPT 3

SEPARATE COMPOSITE SPAR CAP

Figure 34. Spar Cap - Concept Alternatives 1-3

A typical chordwise splice is illustrated in Figure 37. As shown, the splice is of the double shear type consisting of a titanium outer plate and a titanium inner plate, or rib cap. The end of each composite box cover is recessed to accommodate the outer splice plate, and is provided with a titanium insert to maximize bearing strength for the splice fasteners. The hat-section stiffeners of the cover are tapered and terminated at the splice. The outer splice plate is provided with a series of spanwise, machined grooves serving as slot ducts, these being located to coincide with the slot ducts in the box covers. To achieve an efficient LFC system, fastener heads are not permitted to be exposed to the airstream. The countersunk fasteners are installed flush with the outer surface of the splice plate. After completion of the splice, a thin titanium sheet is bonded to the outer surface of the splice. After bonding is complete, LFC slots are sawed in the sheet to align with the slots in the surfaces.

It should be noted that at the wing root, and at Wing Station 455, there is an abrupt change of wing contour which results in a crease along the center of each splice plate.

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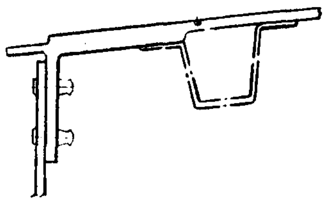
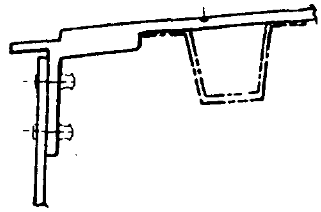


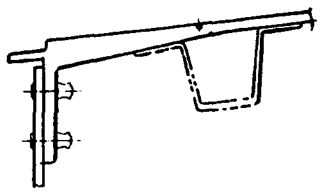
CONCEPT 4

SPAR CAP INTEGRALLY
MOLDED WITH COVER



CONCEPT 5

BLOCK TYPE SPAR CAP
INTEGRALLY MOLDED WITH COVER



CONCEPT 6

TAPERED SPAR CAP INTEGRALLY
MOLDED WITH COVER

Figure 35. Spar Cap - Concept Alternatives 4-6

EVALUATION CRITERIA	CONCEPTS (SCORE 0 - 10) (RANK 1 - 5)											
	1		2		3		4		5		6	
	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE
1. SUCTION DUCT EFFICIENCY	LESS THAN 1% DIFFERENCE											
2. WEIGHT	4	5	6	1	5	3	1	10	3	7	2	8
3. COST	4	5	6	3	5	3	3	6	2	7	1	8
4. INTEGRITY	3	7	5	5	4	6	1	10	6	2	2	9
5. MANUFACTURING	6	2	5	2	4	3	2	7	3	7	1	3
6. MAINTAINABILITY	4	4	5	3	6	2	3	5	2	7	1	10
7. DESIGN FEASIBILITY	6	3	4	5	5	5	2	9	3	6	1	10
TOTAL SCORE	27	26	31	19	29	22	12	47	19	36	8	53
RANK	4	4	6	6	5	5	2	2	3	3	1	1

Figure 36. Evaluation of Spar Cap Concepts

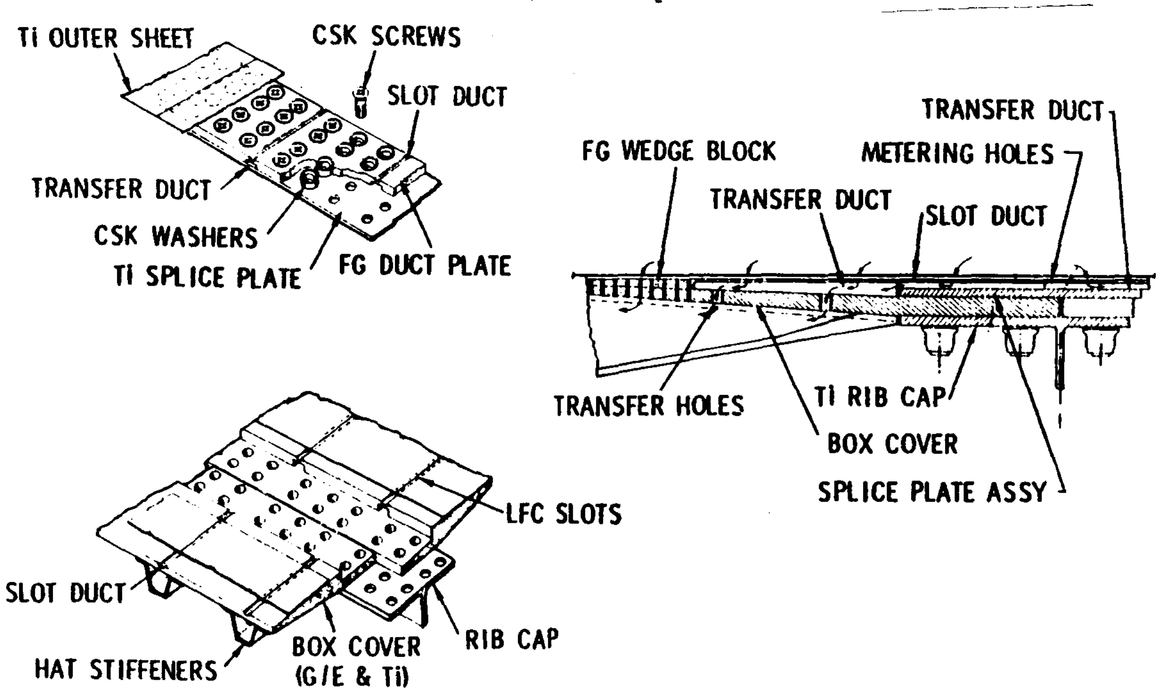


Figure 37. Typical Chordwise Splice

The seven concepts which were considered during the evaluation are depicted in Figures 38, 39, and 40.

Concept 1 - Shown in Figure 38, the boundary layer air passes through the slot ducts and metering holes of the outer splice plate and into transfer ducts formed by machined grooves in the upper surface of the box cover titanium insert. From these ducts, the air is drawn through holes in the insert into the hat-section stiffeners located beneath and, hence, to its point of evacuation. Section B-B of Figure 38 shows the relationship of the splice plate, slot duct, and the transfer duct machined in the cover insert. It is desirable that, in the area of box cover outboard of the recess, the flow of air into the cover slot duct be isolated from the air flow in the transfer duct to avoid flow mixing. A barrier must therefore be provided to isolate these two ducts. Section A-A of Figure 38 shows the graphite-epoxy plies, outside of the titanium insert, to be extended to the recess thus serving as the isolation barrier. This concept may present some difficulties to manufacture because the graphite-epoxy plies located immediately above the machined groove transfer duct cannot be easily pressurized for curing of the laminates. Two possible solutions to this problem are:

(1) Provide a removable plug to fill the groove while curing the laminates and remove the plug after curing.

(2) Cut away the composite laminates in the affected area and replace with a titanium shim of the same thickness.

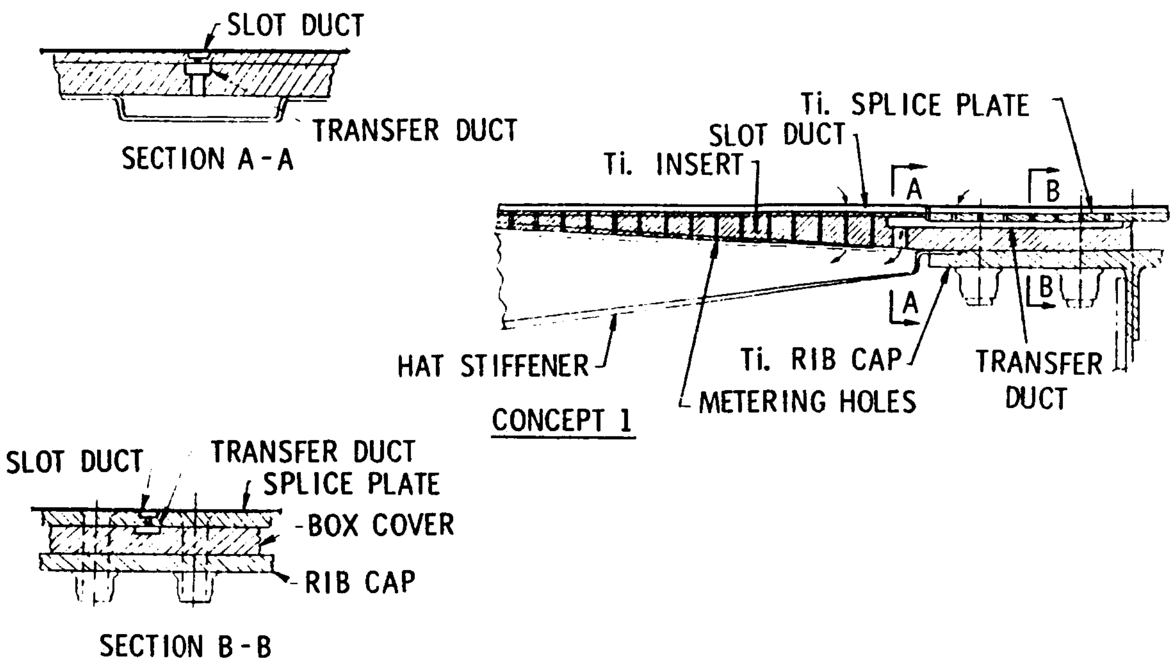


Figure 38. Chordwise Splice - Concept Alternative 1

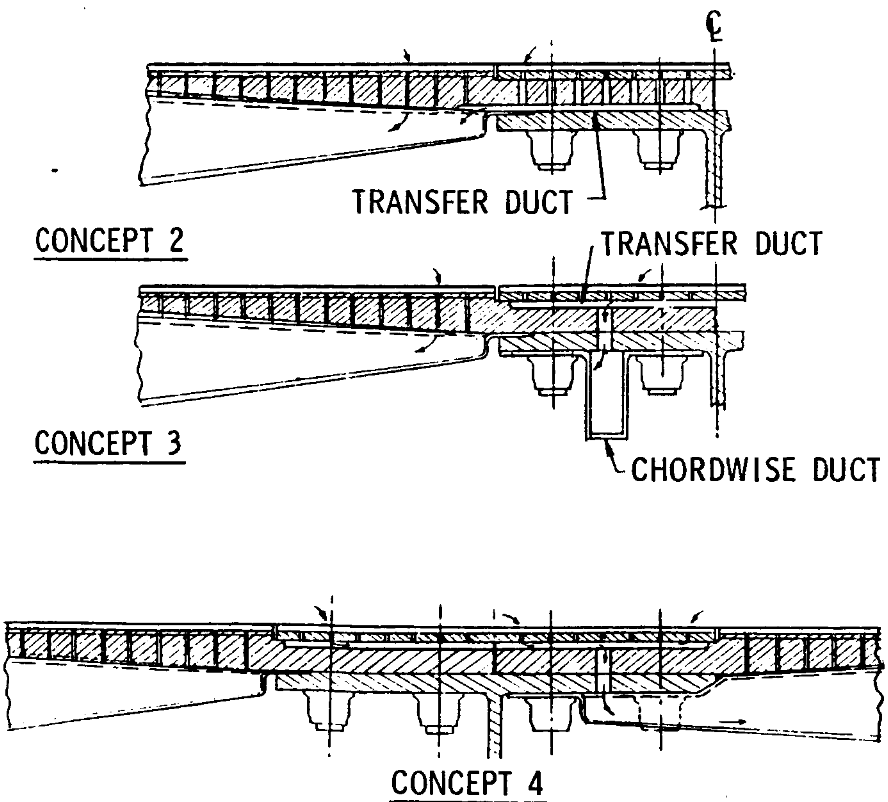


Figure 39. Chordwise Splice - Concept Alternatives 2-4

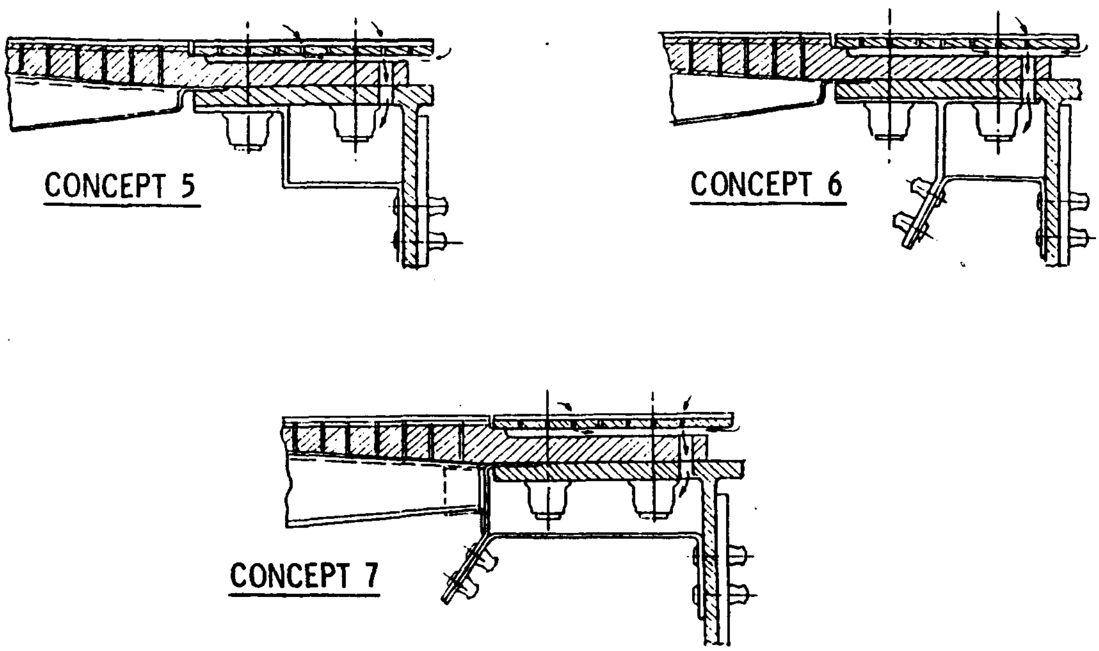


Figure 40. Chordwise Splice - Concept Alternatives 5-7

Concept 2 - Shown in Figure 39 the transfer duct is located on the inner surface of the box cover titanium insert. Boundary layer air is passed through the slot ducts and metering holes of the outer splice plate through larger metering holes in the insert then into the transfer ducts, machined into the insert lower surface, and then directly into the hat stiffeners to the point of evacuation. This concept eliminates the barrier/curing problem of Concept 1, but it requires fifteen metering holes in the box cover insert. A possible problem concerns face-surface-sealing of cover insert to rib cap. Excess sealant could block the transfer duct.

Concept 3 - Shown in Figure 39, the LFC suction provisions for the splice are supplied by two chordwise ducts located on the inner surface of the rib cap. Suction air passes through the outer splice plate into transfer ducts, formed by machined grooves in the cover titanium insert. Holes drilled through the insert and the rib cap allow the air to be drawn into the chordwise duct which then transfers it through the front spar and into the leading edge cavity and hence to its point of evacuation. The hat shaped ducts taper in height from front to rear and are bonded to the titanium rib cap to minimize fuel leakage problems.

Some possible problems associated with this concept are:

(1) A hole is required in the front spar to permit duct air to be drawn into the leading edge cavity, but the high load conditions in the spar cap and cap splice make it undesirable to locate the hole in this region.

(2) The leading edge cavity is divided into two major trunk ducts. The forward duct accommodates suction requirements for the entire upper surface, and a pipe must be provided to pass air from the manifold to this forward cavity of the leading edge.

(3) The installation of splice bolts will be restricted by the height of the duct.

(4) It will be very difficult to match the air flows of the cover slots to the corresponding slots in the splice plate.

Concept 4 - Shown in Figure 39, the joint is asymmetric about the rib centerline. All the suction for the full width of the splice is accommodated by the right hand side hat stiffeners. The box covers and titanium insert are configured similar to those described for Concept 3. The rib cap is bonded to the right hand cover assembly followed by the bonding of the stiffeners joggled over the rib cap flange. This allows direct passage of air from the splices through the cap into the hat stiffener. The hat stiffener is tapered in width to permit its termination end to be located between the splice fasteners. The left side is configured in similar manner as shown for Concept 3 and the stiffener width can be constant if so desired. This concept solves the barrier/curing problems of Concept 1 and the face sealing problems of Concept 2, however, the asymmetric nature of the joint may require additional testing compared to a symmetrical joint. The tapering stiffener width may also be a problem for tooling and fabrication.

Concept 5 - Shown in Figure 40 is a variation of Concept 3 and features a single, removable duct to transfer air from splice to leading edge. The middle row of splice bolts is first installed. The duct is then installed using the outer row of splice bolts and a double row of fasteners attaching the duct to the rib. This concept has the same disadvantages as described for Concept 3 as well as possible fuel leak problems associated with the single row of fasteners in the duct to the rib cap flange.

Concept 6 - Shown in Figure 40 is a variation of Concept 5. It features a two piece duct consisting of a tee bonded to the rib cap, and a channel attached to the tee and rib with a double row of fasteners. This concept minimizes the fuel leak problems of Concept 5; however, it suffers all of the disadvantages described for Concept 3.

Concept 7 - Shown in Figure 40 is a variation of Concept 6 in which the duct vertical duct leg is integral with the box cover which permits an alternative method of terminating the hat stiffeners. A removable channel, attached with a double row of fasteners, to the duct leg and to the rib forms the duct closure. This concept offers a dry bay area for the splice bolts which may be attractive from a fuel leak standpoint, however, the concept still suffers the same disadvantages of Concept 3 and in addition there may be tolerance build up problems associated with the duct closure channel width.

The results of screening are shown in Figure 41 which lists the scoring and rank for each discipline. The final scores are shown on the bottom line of the Figure. Concepts No. 1 and 2 had scores of 51 and 52 respectively and were judged to be equal on score basis. Concept No. 1 was selected as the baseline concept based upon its No. 1 rank positions. Concepts No. 5 through 7 received low scores because of potential pressure differentials at the slot duct.

EVALUATION CRITERIA	CONCEPTS (SCORE 0 - 10) (RANK 1 - 7)													
	1		2		3		4		5		6		7	
	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE
1. SUCTION DUCT EFFICIENCY	1	10	3	9	4	0	2	8	4	0	4	0	4	0
2. WEIGHT	1	6	2	6	7	4	3	5	4	4	5	4	6	4
3. COST	2	5	3	4	4	3	1	6	5	4	6	3	7	2
4. INTEGRITY	2	8	1	10	4	4	3	5	5	3	6	3	7	3
5. MANUFACTURING	1	8	2	6	3	5	4	4	5	4	7	4	6	4
6. MAINTAINABILITY	4	4	3	7	1	8	2	7	6	4	5	4	7	5
7. DESIGN FEASIBILITY	2	8	1	10	4	5	3	6	6	5	5	5	7	5
TOTAL SCORE	13	51	15	52	27	29	18	41	35	24	38	23	44	21
RANK	1	2	2	1	4	4	3	3	5	5	6	6	7	7

Figure 41. Concept Selection - Chordwise Splice

4.2.4 Wing Spar Web Stiffener

Seven spar web concepts depicted in Figures 42 and 43 and three alternatives were investigated.

The front spar web acts as a wall for the main trunk line which collects the suction flow from the lower surface of the wing box. Vertical stiffeners are located inside the wing box on the aft side of the front spar web to provide minimum restrictions for the flow inside the main duct. Concepts 1 through 6 and three alternatives of these six concepts are vertical stiffeners on the aft side of the web. Concept 7 utilizes spanwise stiffeners on the forward side of the web to stiffen the web and to provide a lower restriction for the suction flow than would occur with vertical stiffeners on the forward side of the web.

The following general information was supplied to aid in the evaluation:

- o The spar extends from Box Sta 134 to Box Sta 1570 for a total length of 1436 inches. The spar height at the root is 42 inches and tapers to 28 inches at Box Sta 496 and then to 12.5 inches at the wing tip.

- o The spar web is attached to the spar cap with mechanical fasteners.

Concept 1 - Bonded "J" - In this concept the "J" cross section stiffener is fabricated from 0° and $\pm 45^{\circ}$ graphite epoxy plies. The stiffener is precured and bonded to the spar web.

Concept 2 - Integrally Molded "J" - This concept is similar to Concept 1 except that the stiffener is integrally molded and cocured with the spar web. The $\pm 45^{\circ}$ plies on one side of the spar web are folded to form the leg and flange elements of the stiffener. For this concept the need for a stiffener-to-web-attachment-flange is eliminated and weight is reduced.

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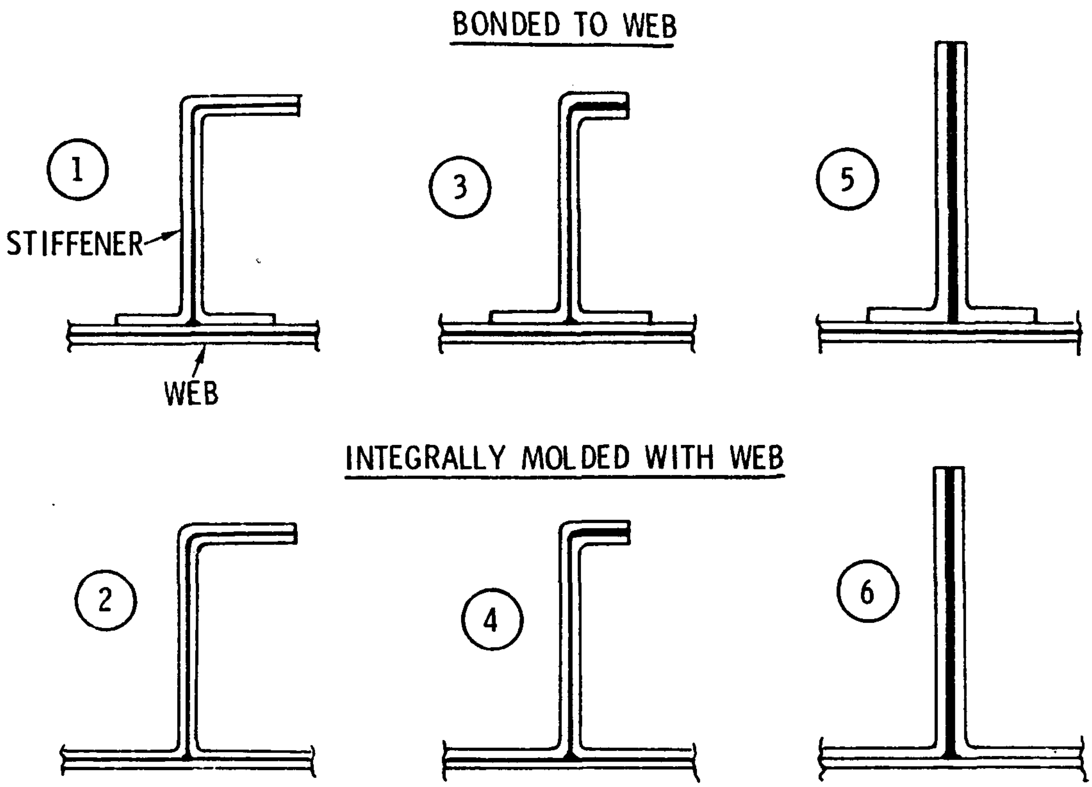

Figure 42. Spar Web Stiffener - Concept Alternatives

Concept 3 - Bonded "J" Small Inner Flange - This concept is similar to Concept 1 except that the inner flange is reduced in width and increased in thickness by the inclusion of additional 0° plies in the flange. In this concept some reduction of weight is achieved by reducing the volume of $\pm 45^{\circ}$ plies at the inner flange.

Concept 3A - Cocured "J" Small Inner Flange - Alternative to Concept 3 in which stiffeners are cocured with the spar web.

Concept 4 - Integrally Molded "J" Small Inner Flange - This concept is similar to Concept 3 except that the stiffener is integrally molded and cocured with the spar web in similar manner to Concept 2.

Concept 5 - Bonded Blade - In this concept the 'blade' type stiffener is fabricated from 0° and $\pm 45^{\circ}$ plies. It can be precured and bonded to the spar web. The blade type stiffener is taller and heavier than for the previous concepts; however, it may be more easily manufactured.

Concept 5A - Cocured Blade - Alternative to Concept 5 to which stiffeners are cocured with the spar web.

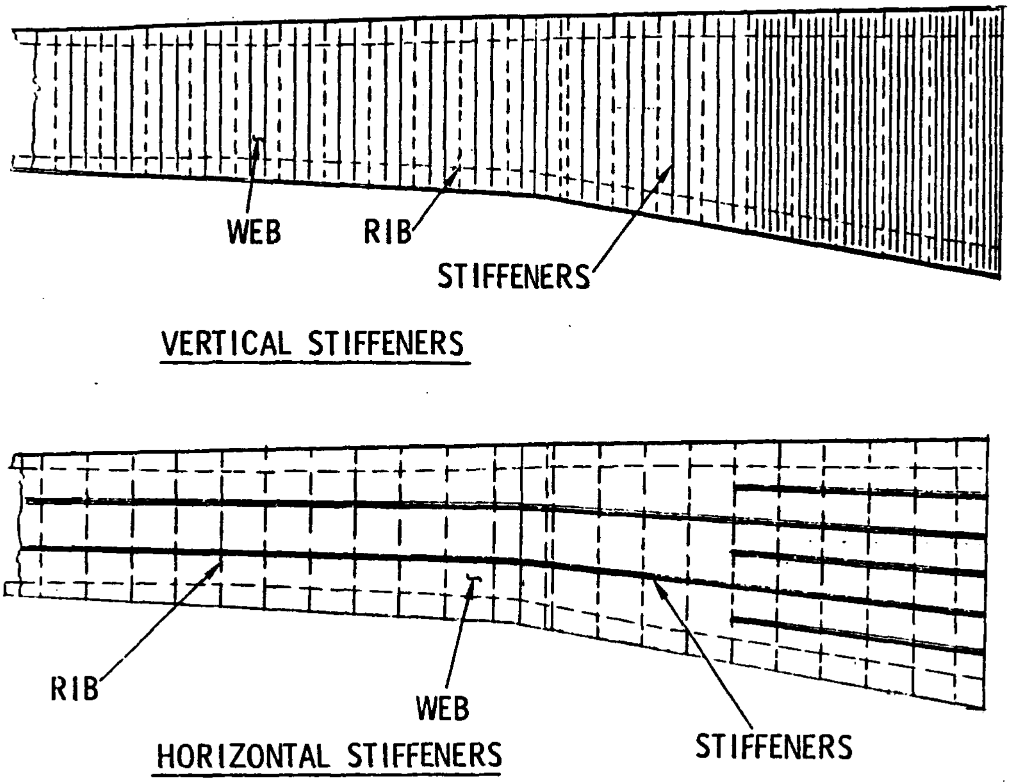

Figure 43. Spar Web Stiffening - Vertical vs Horizontal

Concept 6 - Integrally Molded Blade - This concept is similar to Concept 5 except that the stiffener is integrally molded with the spar web. The 45° plies on one side of the spar web are folded to form the leg elements of the stiffener blade. As the thickness of 45° plies in the web are insufficient to meet the blade requirements additional 45° plies are located in the blade. For this concept the need for a stiffener to web attachment flange is eliminated and weight is reduced.

Concept 7 - Spanwise Stiffeners - Bonded "J" - In this concept the spar web stiffener is oriented spanwise instead of vertical. The "J" cross-section stiffener is fabricated from 0° and $\pm 45°$ graphite epoxy plies. The spanwise stiffener are precured and bonded to the forward side of the spar web. Since the bonded "J" was the most promising for vertical concepts, it was the only stiffener considered for the spanwise concept.

The total rank scores for the 10 spar web stiffener concepts are shown in Figure 44. The spanwise bonded "J" concept, No. 7, received the highest score and the best ranking and was the baseline "selected concept."

EVALUATION CRITERIA	CONCEPTS (SCORE 0 - 10) (RANK 1 - 10)																			
	1		1A		2		3		3A		4		5		5A		6		7	
	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE	RANK	SCORE
1. SUCTION DUCT EFFICIENCY	1	10	1	10	1	10	1	10	1	10	1	10	1	10	1	10	1	10	10	9
2. WEIGHT	7	4	8	4	3	5	5	5	6	5	1	6	9	4	10	4	4	5	2	6
3. COST	9	2	6	4	4	4	10	2	7	4	5	4	8	3	3	6	2	6	1	7
4. INTEGRITY	1	10	2	9	4	9	5	8	6	8	7	7	8	7	9	6	10	5	3	9
5. MANUFACTURING	4	6	7	5	10	1	3	7	6	5	9	1	2	7	5	6	8	4	1	9
6. MAINTAINABILITY	9	1	9	1	8	3	6	4	6	4	5	5	3	6	3	6	2	7	1	10
7. DESIGN FEASIBILITY	4	5	7	5	10	4	2	7	5	5	8	5	3	6	6	5	9	5	1	10
TOTAL SCORE	35	38	40	38	40	36	32	43	37	41	36	38	34	43	37	41	36	42	19	60
RANK	4	7	9	7	9	10	2	2	7	6	5	7	3	2	7	2	5	5	1	1

Figure 44. Evaluation of Spar Web Stiffener Concepts

4.3 MATERIAL SELECTION AND EVALUATION

Development work at Lockheed and elsewhere has shown that graphite-epoxy materials will produce significant improvements in performance and weight reductions in transport aircraft that may become operational in the 1990 time period. In particular, the T300/5208 high resin graphite-epoxy material has been successfully used in LFC panel skins and hat-section stiffeners in Phase I of the LFC development program, as well as in other advanced composite structures. Prior to initiation of this program, the basic T300/5208 high resin material was revised to T300/5208DV, which was a devolitized version with more precise advancement as well as lower resin content. The revised formulation was supposed to resolve processing problems experienced by the aerospace industry with the basic T300/5208 material. After processing problems with the T300/5208DV material, the Hercules 350°F-curing AS4/3502 graphite-epoxy material was evaluated. The AS4/3502 material was tested to the requirements of General Dynamics Specification FMS 2023. After the first order of AS4/3502 material was processed, the areal weight and resin content were changed in the subsequent purchase orders to satisfy processing requirements for the LFC surface panels. The AS4/3502 material will conform with Lockheed Specification C-22-1379/114 with the exception of areal weight and resin content.

In applications which require a mixture of graphite-epoxy tape and graphite-epoxy fabric, a 350°F-curing fabric is required. Graphite-epoxy tape material with a 250°F cure is not used on this program because of the generally poor hot-wet properties. The Fiberite 350°F-curing HMF 133/34 graphite-epoxy fabric procured to the requirements of U.S. Navy Specification NAVORD-WS16042 was used in this program since its resin system is compatible with both AS4/3502 and T300/5208 350°F-curing systems. The application in the program having mixed graphite-epoxy tape and fabric materials is the rib-cap duct cover. Thus, the Fiberite HMF 133/34 graphite-epoxy fabric material whose allowable strengths are comparable to T300/5208 fabric material will be used for 350°F fabric material applications in the remainder of the program.

Applications of 250°F-curing graphite-epoxy materials are required in several components of the LFC wing surface panel. The resin system of the Hercules 250°F-curing A370-5H-1908 graphite-epoxy fabric material is compatible with the AS4/3502 graphite-epoxy tape material, and it has been used successfully in fabrication of the ancillary test specimens. The A370-5H/1908 graphite-epoxy fabric was selected for use in 250°F applications in the remainder of the program. Other potential candidates that were not evaluated, but are considered as alternates for 250°F-curing fabric applications, are T300/5225 (Narmco), F155 (Hexcel) and BP919 (American Cynamid) fabric material.

Table 2 summarizes graphite-epoxy material applications:

The LFC wing surface concept used in phase I programs conducted at Lockheed utilized an outer surface sheet in which LFC suction slots are sawed. This outer surface sheet material was 6AL-4V annealed titanium whose thickness was approximately 0.020 inch. The same concept has been used in the design and fabrication of specimens in this program. Therefore, the primary candidate material for the outer surface sheet is 6AL-4V annealed material.

Other applications in fabrication of the LFC wing surface which require the use of titanium are shims for interleafing graphite-epoxy laminates in splice joint areas for increasing fatigue strengths in the mechanically fastened splice plates. The 6AL-4V annealed titanium sheet and plate procured to the requirements of MIL-T-9046 specification was used successfully in fabrication of the ancillary test specimens. Thus, the 6AL-4V annealed sheet and plate material was selected for manufacturing.

The FM73 film adhesive has been used successfully for bonding the graphite-epoxy hat-section stiffeners to graphite-epoxy LFC surface panel skins and it is considered as the primary candidate for this application in fabrication of the LFC specimens in this program. The EA9628 and AF163-2 film adhesives were evaluated coincident with FM73 in a series of material verification tests. No definitive conclusions could be drawn from these tests. However, the FM73 adhesive was selected for bonding the hat-section stiffeners to the surface skin laminates based on its successful performance in delaminating the graphite-epoxy laminates in lieu of bondline failures. The weight of the FM73 adhesive is 0.06 lb/ft^2 (PSF) and it may be procured to the requirements of specification MMM-A-132.

APPLICATION	PRIMARY MATERIAL SELECTION	ALTERNATE MATERIAL SELECTION
350°F-CURING TAPE MATERIAL	AS4/3502	T300/5208
350°F-CURING FABRIC MATERIAL	HMF 133/34	T300/5208
250°F-CURING FABRIC MATERIAL	A370-5H/1908	T300/5225, F155 & BP919

TABLE 2.
SUMMARY GRAPHITE/EPOXY MATERIALS APPLICATIONS

The FM123-4 film adhesive in the 0.045 PSF weight has been used successfully in bonding the exterior titanium sheet to the graphite-epoxy surface skin laminate in Phase I of the LFC development program. In the interest of minimizing thermal stresses in the titanium sheet to surface skin laminate bond and of minimizing flow of the adhesive in the slot-duct region during the cure cycle, two candidate adhesives were evaluated. The FM123-4 and EA9601.2 film adhesives were evaluated and each adhesive was cured at both 180°F and 200°F. Both adhesives were evaluated through the fabrication of single lap-shear specimens and process control panels. The lap shear specimens were static tested and the process control panels were sectioned to evaluate the adhesive flow characteristics. The evaluation resulted in selection of the FM123-4 adhesive procured to the requirements of Lockheed Specification STM30-102 and cured at 200°F. This selection was based on the following:

(1) Flow characteristics of the adhesive in the slot duct region during the cure cycle.

(2) Adhesive bondline thickness variability.

(3) Adhesive bond strength.

(4) Bondline failure mode. Graphite-epoxy laminate delamination is desired.

Subsequent to the above described evaluation, FM123-2 film adhesive was used in the fabrication of several ancillary test specimens. The FM123-2 adhesive has the same formulation as the FM123-4 adhesive but it is on a mat carrier. Since FM123-2 adhesive exhibits characteristics that are comparable to FM123-4 adhesive, it was selected as an alternate for the FM123-4 for bonding the titanium sheet to the graphite-epoxy surface skin laminate.

A specific requirement exists for bonding an exterior titanium sheet to the wing chordwise joint region. The FM123-4 or FM123-2 film adhesive was selected for this application. The adhesive bond will be accomplished with the aid of portable heaters.

A thin film adhesive is required for bonding the titanium sheets forming the laminate for splicing the LFC wing surface structure. A similar requirement exists for bonding titanium sheet interleaves to uncured graphite-epoxy tape lay-ups in the fabrication of transition regions of LFC wing surface structures in which mechanically fastened joints are required. The AF147U adhesive (3M Company) having a weight of 0.03 PSF and procured to the requirements of MMM-A-132 specification has been used successfully in these applications in the fabrication of ancillary concept selection specimens. The FM300 adhesive was selected as an alternate for these applications.

Adhesive materials applications are summarized in Table 3.

APPLICATION	PRIMARY ADHESIVE SELECTION	ALTERNATE ADHESIVE SELECTION
1. GRAPHITE-EPOXY HAT-SECTION STIFFENERS TO GRAPHITE-EPOXY SURFACE SKIN LAMINATE	FM73	EA9628 AND AF163-2
2. EXTERIOR TITANIUM SHEET TO GRAPHITE-EPOXY SURFACE SKIN LAMINATE	FM123-4	FM123-2
3. EXTERIOR TITANIUM SHEET TO WING CHORDWISE JOINT REGIONS	FM123-4	FM123-2
4. TITANIUM SHEETS FOR FORMING THE LAMINATE FOR SPLICING THE LFC WING SURFACE STRUCTURE	AF147U	FM300
5. TITANIUM SHEET INTERLEAVES TO UNCURED GRAPHITE-EPOXY TAPE LAY-UPS IN THE TRANSITION REGIONS OF LFC WING SURFACE STRUCTURES	AF147U	FM300

TABLE 3.
SUMMARY OF ADHESIVE MATERIAL APPLICATIONS

5.0 DESIGN REQUIREMENTS FOR LFC WING SURFACE AND STRUCTURE

5.1 DESIGN STRUCTURAL CRITERIA

The structural design criteria for both the LFC and non-LFC aircraft comply with FAR 25 structural design criteria. A design life goal of 90,000 flight hours is used for the long-range aircraft.

The aircraft are designed for a 2.5g symmetric maneuver load condition in accordance with FAR 25. Advanced technology and materials are applied to both LFC and non-LFC aircraft. Active controls are utilized to reduce wing bending moments. An active gust load alleviation system is used to reduce wing bending due to gust. Preliminary sizing routines use a gust velocity, U_{DE} = 40 fps in lieu of 50 fps. Active flutter suppression is used to provide the FAR 25 (1.2) speed margin above dive speed. A soft landing gear permits the use of a taxi load factor of 1.5g and the use of relaxed static stability permits location of the aft center of gravity limit at 55 percent MAC. A speed-altitude schedule shown in Figure 45 was derived using FAR 25 requirements for a 7.5 degree upset for 20 seconds followed by a 1.5g pull-up. The following weights were used for structural design:

- Gross weight = 592,205 lbs including 218,679 lbs of fuel.
- Maximum zero fuel weight = 373,526 lbs
- Design landing weight = 484,849 lbs

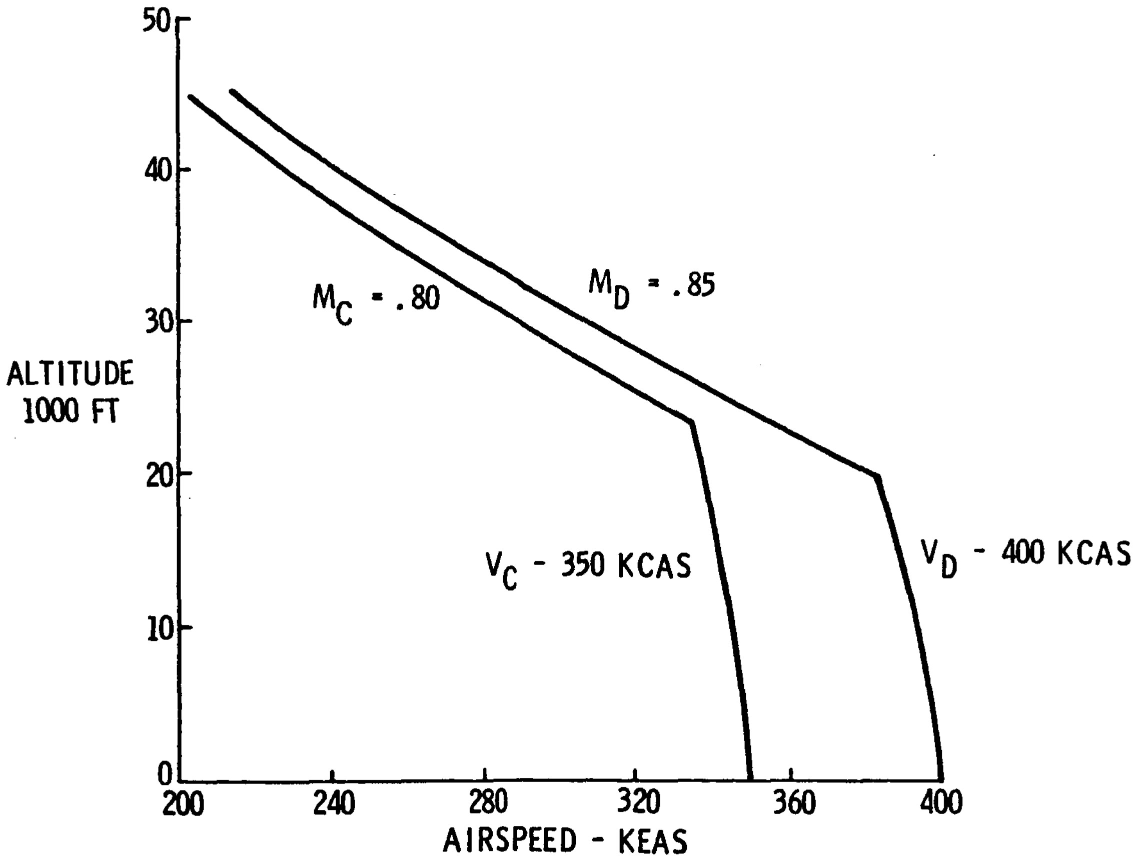

Figure 45. Design Airspeeds

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5.2 DESIGN LOAD CONDITIONS

The aerodynamic configuration from the Phase I studies was used to provide the input data to the structural loads program, ANSWER (Analytical Structural and Weight Estimation Routine) and to the flutter analysis program, ARF (Automated Resizing for Flutter).

The seven critical load conditions for the baseline LFC wing are shown in Table 4.

The wing torsional stiffness distribution required for flutter prevention was determined by means of the flutter optimization computer program ARF. This program operates in conjunction with the structural synthesis program ANSWER to develop minimum weight wing designs which satisfy both strength and flutter speed constraints. Table 5 shows the estimated stiffness requirements.

ARF uses geometric, mass, and available stiffness data computed by ANSWER for an initial strength-optimum design, and performs a wing flutter analysis. If the flutter speed for this design is less than that required, the program computes flutter velocity weight derivatives for each of nine spanwise wing segments and iteratively resizes the wing structure (EI and GJ) to achieve the required flutter speed with a minimum expenditure of weight. The resulting torsional stiffness distribution is finally input as a design requirement to ANSWER to make the final iteration in the aeroelastic loads calculations and strength sizing.

Because the LFC design concept includes active flutter suppression, the wing was structurally sized for an unaugmented flutter speed equal to the design dive speed of 400 KEAS. The required 20 percent flutter safety margin is to be provided by the active flutter suppression system. Analyses and tests of prototype flutter suppression systems, including wind tunnel tests conducted by Lockheed-Georgia Company, indicate that a 20 percent flutter speed increase is achievable with a single-channel active flutter suppression system.

LOAD TYPE	N_z	WEIGHT (LB)	FUEL (LB)	U_{de} (FPS)	VEQ (KTS)	MACH	ALTITUDE 1000'
1. MANEUVER-GROSS	2.5	592 205	218 679	0	400	.604	0
2. POSITIVE GUST	2.258	592 205	218 679	50	350	.78	20
3. POSITIVE GUST	2.679	373 526	0	50	350	.78	20
4. NEGATIVE GUST	- .258	592 205	218 679	-50	350	.78	20
5. NEGATIVE GUST	- .679	373 526	0	-50	350	.78	20
6. TAXI	-1.5	592 205	218 679	-	-	-	0
7. MANEUVER ZFW	2.5	373 526	0	0	400	.604	0

TABLE 4.
LOAD CONDITIONS FOR BASELINE LFC WING

Two fuel conditions were analyzed for flutter; mission fuel (218,680 pounds) and empty. Previous experience indicates that adequate flutter speeds for intermediate fuel conditions can be achieved by the proper design of the fuel system.

The critical wing limit bending moments are shown in Table 6.

The critical estimated wing tip limit deflections for the LFC baseline wing are shown in Table 7. The critical estimated maximum wing panel surface load per inch for the baseline wing is shown in Table 8. The ultimate spar loads shown in Table 9 were taken from the ANSWER output.

SEMISPAN %	EI $LB\text{-}IN^2 \times 10^{-10}$	GJ $LB\text{-}IN^2 \times 10^{-10}$
75	4.7	4.9
55	14.9	15.5
35	34.8	35.0
14	97.0	56.0

TABLE 5.
WING STIFFNESS REQUIREMENTS FOR DESIGN

SEMISPAN %	UP $IN\text{-}LB \times 10^{-7}$	DOWN $IN\text{-}LB \times 10^{-7}$
75	1.0	- .4
55	3.4	-1.6
35	8.0	-4.2
14	16.0	-8.2

TABLE 6.
WING DESIGN LIMIT BENDING MOMENTS

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UP (IN)	DOWN (IN)
182	-91

TABLE 7.
WING TIP DEFLECTION AT LIMIT LOAD

SEMI SPAN %	UPBENDING (KIPS/IN)		DOWNBENDING (KIPS/IN)	
	N_x	N_{xy}	N_x	N_{xy}
75	7.1	1.0	2.6	0.1
55	15.9	1.7•	7.6	0.3
35	25.7	2.4	13.7	0.4
14	29.1	2.8	14.7	1.1

•TEST SECTION

TABLE 8.
WING DESIGN SURFACE LOADS - ULTIMATE

SEMI-SPAN	FRONT SPAR UPPER CAP P_y (K)	FRONT SPAR LOWER CAP P_y (K)	FRONT SPAR WEB SHEAR (K/IN.)	REAR SPAR UPPER CAP P_y (K)	REAR SPAR LOWER CAP P_y (K)	REAR SPAR WEB SHEAR (K/IN.)	CRUSHING LOAD/RIB (LB/IN.)
.95	- 2.76	+ 2.76	.90	- 1.9	+ 1.9	.39	- 5.6
.85	- 21.9	+ 16.8	2.06	- 15.4	+ 15.4	.74	-170.1
.75	- 55.7	+ 55.7	3.05	- 39.2	+ 39.2	1.15	-458.5
.65	-101.4	+101.4	3.88	- 71.3	+ 71.3	1.53	-560.0
.55	-156.4	+156.4	4.61	-110.0	+110.0	1.89	-617.4
.45	-219.0	+219.0	5.26	-154.0	+154.0	2.26	-639.5
.353	-301.2	+301.2	6.51	-212.0	+212.0	2.63	-693.7
.253	-332.1	+332.1	4.76	-282.0	+282.0	1.10	-641.9
.145	-327.4	-165.0	8.49	-349.0	+349.0	6.30	-653.5

TABLE 9.
SPAR ULTIMATE LOADS

5.3 ENVIRONMENT REQUIREMENTS

The baseline LFC wing surface panels were designed for:

- o Temperature range for -65° to $160^{\circ}F$
- o Foreign object damage consideration
- o Lightning strike
- o Corrosion environment

  - fuels
  - lubricants
  - oils
  - cleaning fluids
  - anti-ice fluids
  - sand/rain
  - hydraulic fluids

This environment is representative of those encountered in airline operation.

5.4 GENERAL REQUIREMENTS

Special emphasis was placed on consideration for producibility, maintenance, inspection, and repair of the LFC wing surface structure and associated LFC systems on the wing surface of the 1993 transport aircraft. Emphasis was placed on maintaining the wing surface for LFC and not on structural repair.

To the extent possible, fabrication and assembly techniques and inspection methods demonstrated during this development program are compatible with existing customer maintenance practices, support equipment, and personnel skill levels. Specific equipment needed for use in in-service inspection are identified.

5.5 LFC SUCTION SYSTEM REQUIREMENTS

The LFC suction system was developed during Phase I and reported in Section 6.3 of NASA contractor report CR159253, Reference 1. These criteria are summarized below:

- o Both surfaces are laminarized to 75 percent chord.
- o Suction slots start at 1.2 percent chord on the upper and 1.1 percent chord on the lower surface.
- o Suction is applied to chordwise splices and to access doors.

- o Design Limit Pressures are -3.0 psi (ground checkout) to +2.5 psi (purge).
- o Two washer slots are used on the upper surface and as many as possible on the lower surface.

The slot widths and spacings are tabulated in Table 10 for the upper and lower surfaces.

The dimensions of the slot ducts between 18 and 70 percent chord are shown in Figure 11. Metering holes in the slot ducts are 0.05 in in diameter spaced 0.30 in center-to-center. Internal ducting consists of the spanwise hat ducts and chordwise rib ducts. A 0.50 in diameter metering hole on the forward side of the hat meters the flow to the chordwise rib ducts. The rib ducts are spaced at 70 in, and feed the flow through holes in the front spar to the leading edge collector ducts.

5.6 PRODUCIBILITY DESIGN REQUIREMENTS

For use in estimating production costs, a set of ground rules is defined. These are listed below:

UPPER SURFACE DESIGN			LOWER SURFACE DESIGN		
X/C	SLOT WIDTH (IN.)	SLOT SPACING (IN.)	X/C	SLOT WIDTH (IN.)	SLOT SPACING (IN.)
.012	.0035	.62	.011	.003	.875
.018	.0035	.62	.017	.003	.920
.027	.0050	1.12	.025	.004	1.50
.037	.0050	1.25	.035	.004	1.50
.048	.0050	1.25	.046	.006	2.30
.062	.0050	1.75	.059	.006	2.65
.076	.010	5.00	.073	.006	3.25
.092	.010	5.00	.089	.006	3.25
.100	.010	5.00	.100	.006	3.25
.200	.013	6.00	.200	.015	12.00
.700	.013	6.00	.700	.015	12.00

TABLE 10.
SLOT WIDTHS AND SPACING

- All costs are expressed in January 1, 1981 dollars
- Total Production: 350 units
- Production Span: 10 years
- Production Rate: 1.5 to 4 units per month
- Learning Curves:
  - Labor: 75 percent
  - Material: 96 percent

5.7 SURFACE ACCEPTANCE CRITERIA

The operational acceptance criteria for the surface deformation defects is shown in Table 11. These allowable variations (.030 (10.0) = 0.30 in. height in 10 in. length) were used to evaluate the test panels. Most of the smoothness criteria are based on the NOR 61-141 (Ref 3) (X-21) criteria. The downstep at a slot edge of 0.002 in was selected arbitrarily to account for the increase in the local velocity of the boundary layer above the X-21 design.

DEFECT		PARAMETER	ALLOWABLE VARIATION
WAVINESS		HEIGHT (LENGTH)	.030 (10.0)
STEP, JOINT	- DOWN	HEIGHT	.006
	- UP	HEIGHT	.0137
STEP, SLOT EDGE	- DOWN	HEIGHT	.002
	- UP	HEIGHT	.008
	- DOWN	THICKNESS	.030
GAP, JOINT	- CHORDWISE	WIDTH	.101
	- SPANWISE	WIDTH	.161
GAP, SLOT	- .013 NOMINAL	WIDTH	.0013

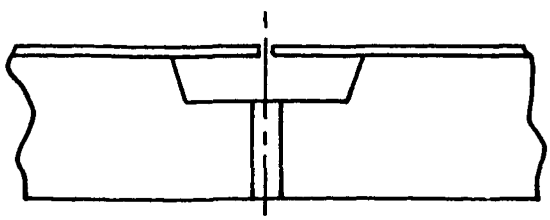

TABLE 11.
SURFACE DEFORMATION DEFECTS

Operational acceptance criteria for the interior deformation defects is listed in Table 12. The metering-hole diameter allowable variation of 0.001 in represents the most difficult tolerance to maintain.

The operational acceptance criteria for the outer surface are shown in Table 13. Allowable cracks in the outer sheet in the slot-duct area are limited to 0.15 in (distance from slot to slot duct wall). Cracks in other areas of the outer sheet are assumed to be within the smoothness step and gap criteria, and no other limitations are imposed. Allowable delamination of the outer sheet is limited to 0.25 in diameter.

DEFECT	PARAMETER	ALLOWABLE VARIATION
SLOT DUCT - .08/.32	HEIGHT/WIDTH	.004/.016
TRANSFER DUCT - 2.0/2.0	HEIGHT/WIDTH	.1
METERING HOLE - .052	DIAMETER	.001

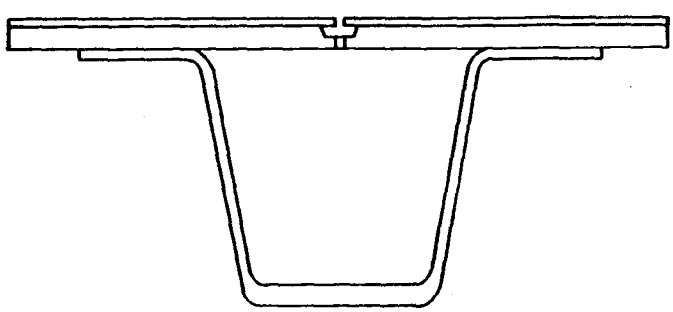

TABLE 12.
INTERIOR DEFORMATION DEFECTS

DEFECT	PARAMETER	ALLOWABLE VARIATION
CRACK		
OUTER SHEET - DUCT AREA	LENGTH	.15
- BOND AREA	LENGTH	NO LIMIT
DELAMINATION		
OUTER SHEET - BOND AREA	DIAMETER	.25

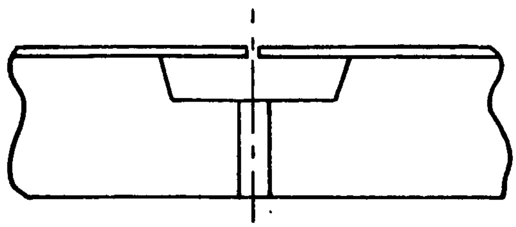

TABLE 13.
OUTER SURFACE - DEFECTS

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6.0 DEVELOPMENT TEST PROGRAM AND SPECIMENS

The overall plan for developing the LFC wing surface is summarized in Figure 46. A series of material verification (MV) specimens were fabricated and tested to verify the materials and processes. Concept selection (CS) specimens were used to develop the critical design details, tools and manufacturing processes. Concept verification (CV) will be accomplished by the manufacture and test of the large surface panel. The first panels made on the wing surface tool will be structurally tested to verify the design and the tool. These panels are identified as CV-1 and CV-2. The large panel assembly with ribs, rib ducts, chordwise joints and spar caps is identified as CV-4. CV-4 will be installed in a simulated wing box for testing the integrated LFC wing structure under simulated flight loads. The CV panels were not fabricated in the original WSSD program, but it is planned to fabricate some of these panels in a later program.

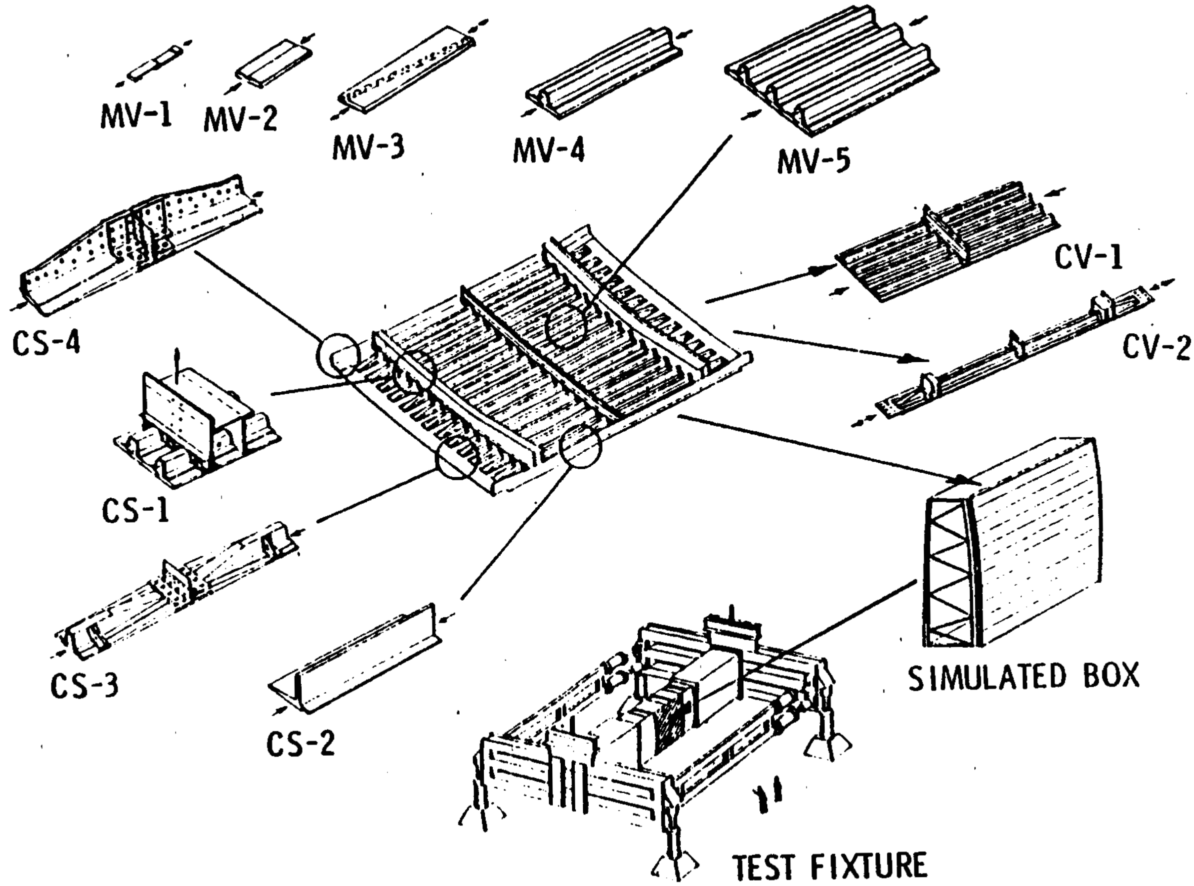

Figure 46. Integrated Wing Surface Panel Development

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6.1 MVI SPECIMEN

Thirty lap-shear specimens, designated MV1, were used to screen, evaluate and select the adhesives and bonding temperatures. The MV1 test specimen is shown in Figure 47. Photographs shown in Figure 48 show ten failed MV1 specimens that were tested at room temperature. The ten specimens were fabricated with FM123-4 adhesive of which five specimens were cured at 180°F and five specimens were cured at 200°F.

6.2 MV2 SPECIMEN

Twenty-four specimens, designated MV2, and shown in Figure 49, were used to evaluate the slot ducts, metering holes, and bond of titanium sheet to the graphite/epoxy surface. In particular, a minimum flow adhesive was desired to prevent the accumulation of adhesive in the slot ducts. The MV2 specimen is shown in Figure 50.

6.3 MV3 SPECIMEN

MV3 was designed to evaluate the effect of combined axial and shear loads in the wing surface with slot ducts with closely spaced metering holes.

The photographs in Figure 51 are close-up and distant views of the MV3 specimen installed in the testing machine. In the close-up view on the left, the specimen test section, including the slanted slot duct with the metering holes, is clearly visible. In the view on the right, the entire MV3 specimen is shown in the test machine with accompanying instrumentation and load control and monitoring equipment.

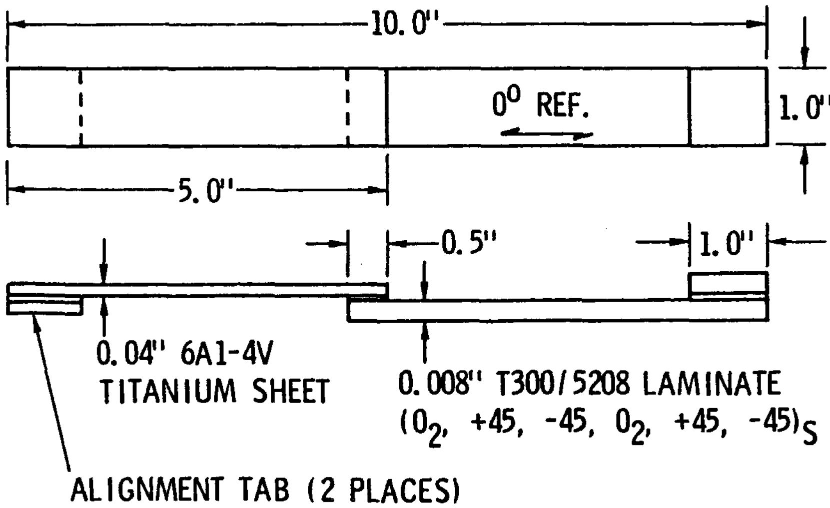

Figure 47. Material Verification Test Specimen - MV1

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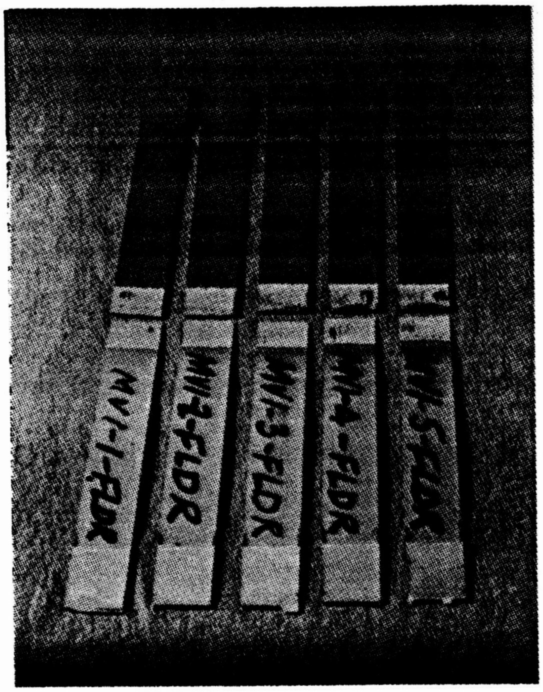

FM 123-4
CURED AT 180°F
TESTED DRY AT RT

FM 123-4
CURED AT 200°F
TESTED DRY AT RT

Figure 48. Failed MV1 Specimens - FM123-4 Adhesive (Room Temperature)

- OBJECTIVE: EVALUATE PROCESSES FOR FABRICATION OF G/E LAMINATES, FORMING LFC SLOT DUCTS, AND DRILLING SLOT DUCT METERING HOLES
- MATERIALS: T300/5208 G/E TAPE
  AS4/3502 G/E TAPE
- COMPRESSION TESTS
- ENVIRONMENTAL CONDITIONS
  - ROOM TEMPERATURE
  - CONDITIONED DRY AT -65°F AND TESTED AT -65°F
  - "HOT-WET"
  - "COLD-WET"

Figure 49. MV2 Specimen Test (24 Specimens)

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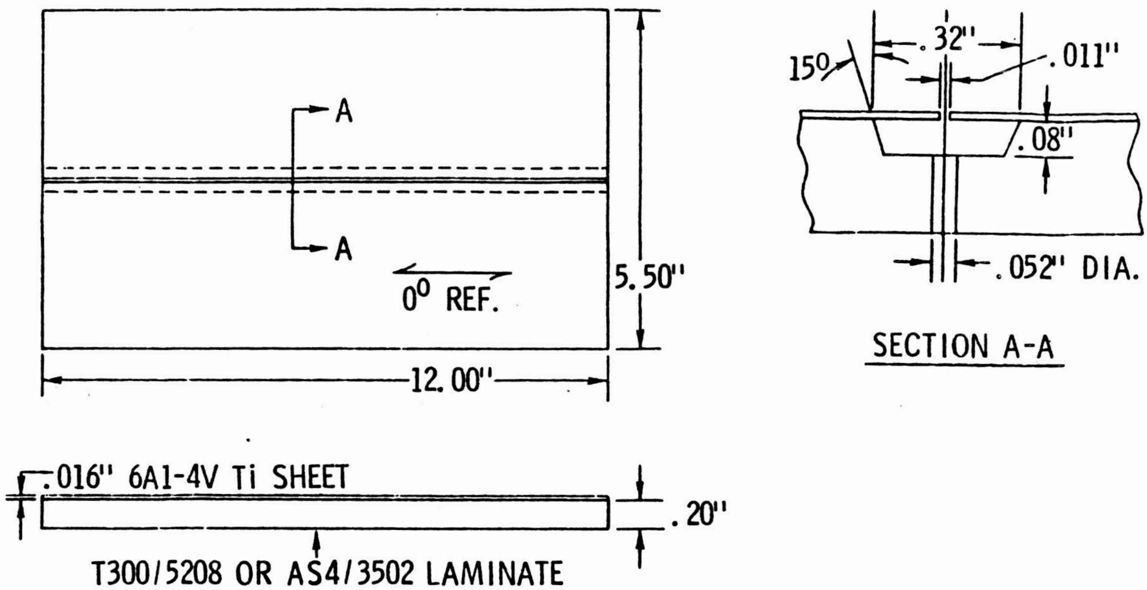
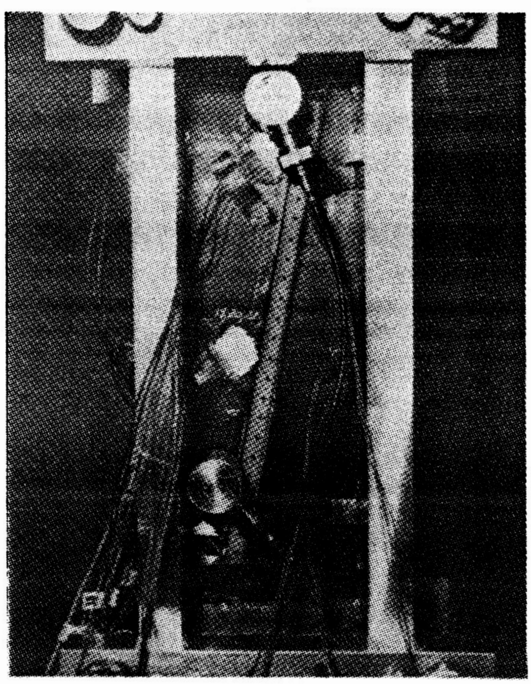
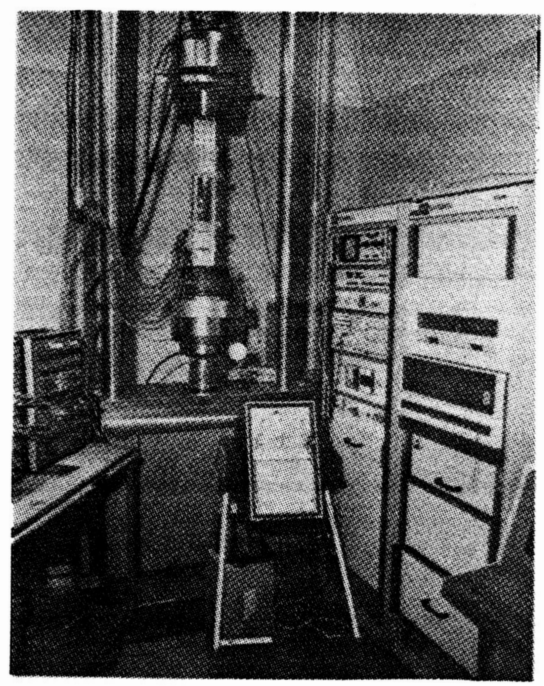


Figure 50. Material Verification Test Specimen - MV2



CLOSE-UP VIEW



DISTANT VIEW

Figure 51. MV3 Specimen Installed in Testing Machine

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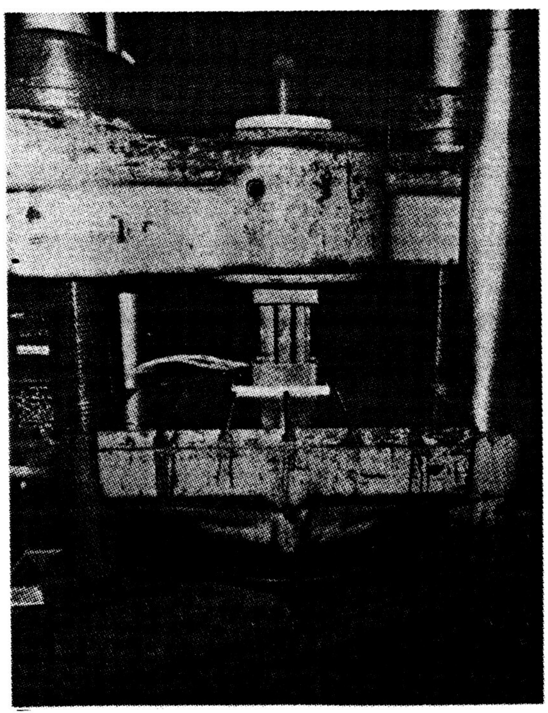
6.4 MV4 SPECIMEN

MV4 was similar to the MV2 specimen with the addition of a hat stiffener. The objective of this test was to evaluate the adhesives and processes for bonding the graphite/epoxy hats to the graphite/epoxy skin.

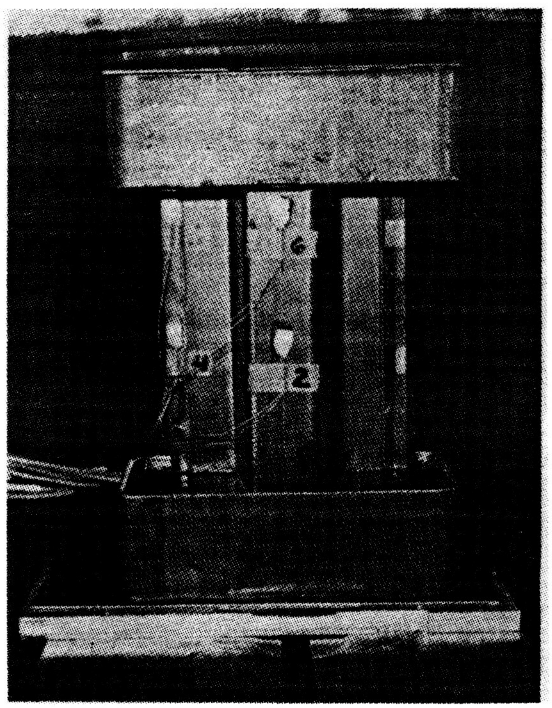
The photographs, shown in Figure 52, are close-up and distant views of a typical MV4 specimen installed in a universal testing machine. The hat-section stiffener side of the specimen is shown in both views. The close-up view on the right shows the specimen instrumentation on the hat-section stiffener side of the specimen, including acoustic emission monitoring instrumentation.

6.5 MV5 SPECIMEN

MV5 was designed to verify the materials and processes selected for the wing surface. The skins and hats were fabricated with Hercules AS4/3502 graphite/epoxy tape. FM73 was used to bond the graphite/epoxy hats to the graphite/epoxy skin. Metering holes were drilled in the slot ducts, and the titanium skins were bonded with FM123-4 adhesive. Slots were cut in the titanium outer skin. The dimensions and configuration of MV5 are shown in Figure 53.



DISTANT VIEW



CLOSE-UP VIEW

Figure 52. Typical MV4 Specimens in Test Machine

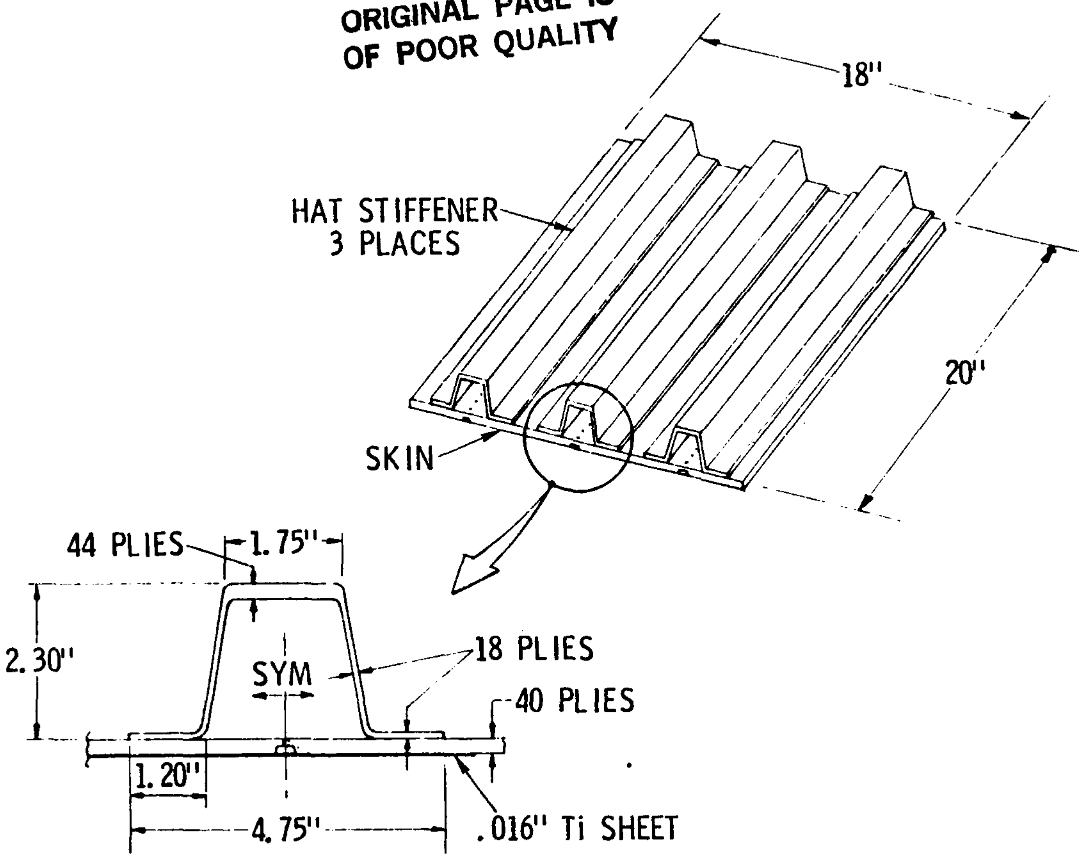


Figure 53. Material Verification Specimen MV5

6.6 CS1 SPECIMEN

CS1 was designed to evaluate the design and producibility of a chordwise rib-cap duct/hat-stiffened LFC wing surface assembly. Details of the CS1 specimen are shown in Figure 54. The photograph, Figure 55, shows the first CS1 specimen that was fabricated. As shown, the specimen is complete with the exception of the rib-cap duct cover which will be mechanically fastened to the rib-cap duct.

6.7 CS2 SPECIMEN

The second group of concept selection specimens was fabricated and tested to evaluate the spar cap design and spar cap to spar web joint. Figure 56 shows the CS2 concept selection specimen. A cross section of the specimen is presented which shows the graphite/epoxy ply orientations. Figure 57 shows the spar cap leg to spar web joint. This thinner member in the joint specimen represents the spar web while the thicker member represents the spar cap leg.

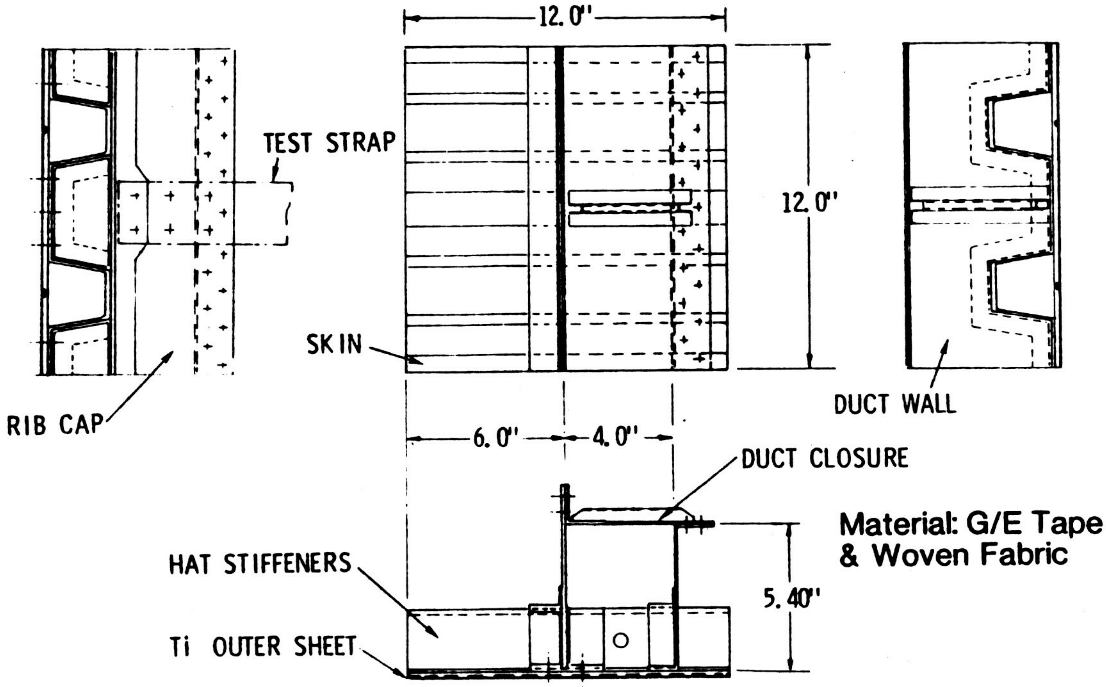


Figure 54. Concept Selection Specimen CS1

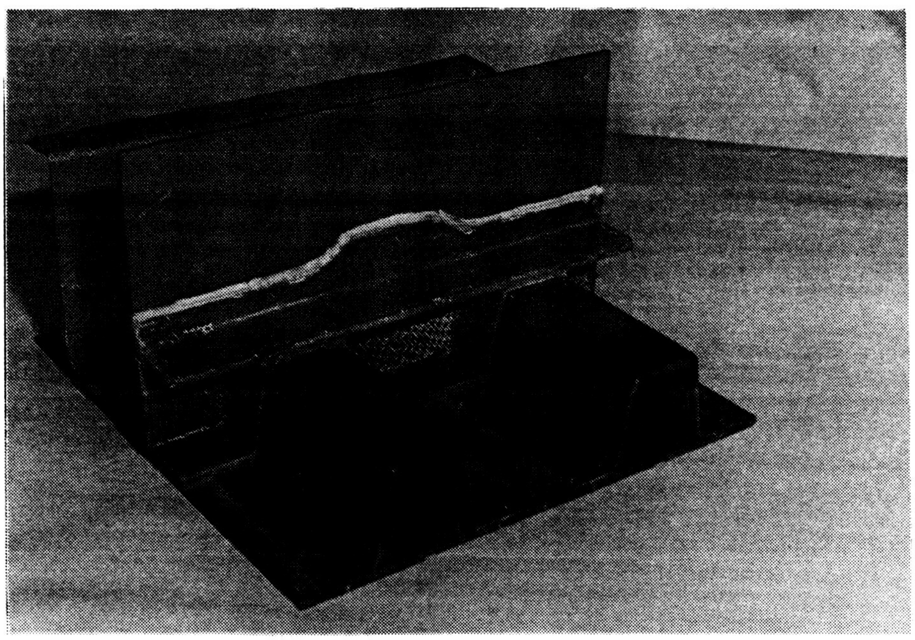


Figure 55. CS1-1 Test Specimen Less Rib Cap Duct Cover

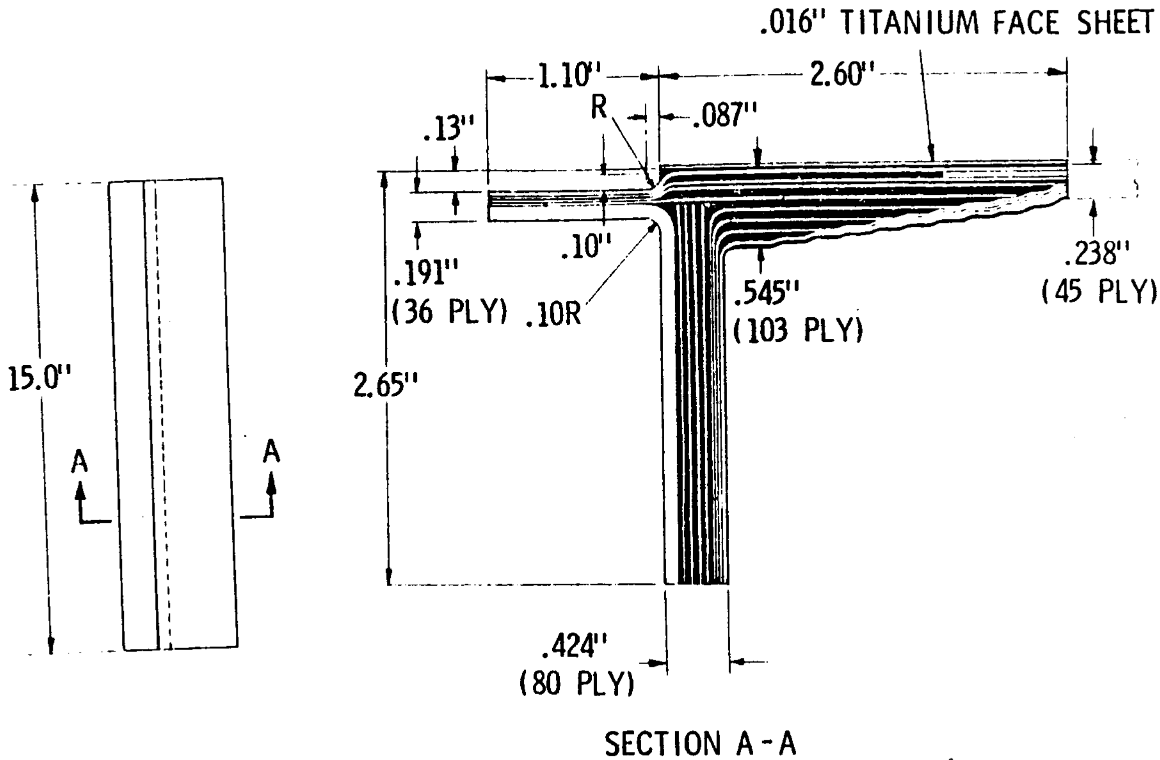


Figure 56. Concept Selection Specimen CS2-1

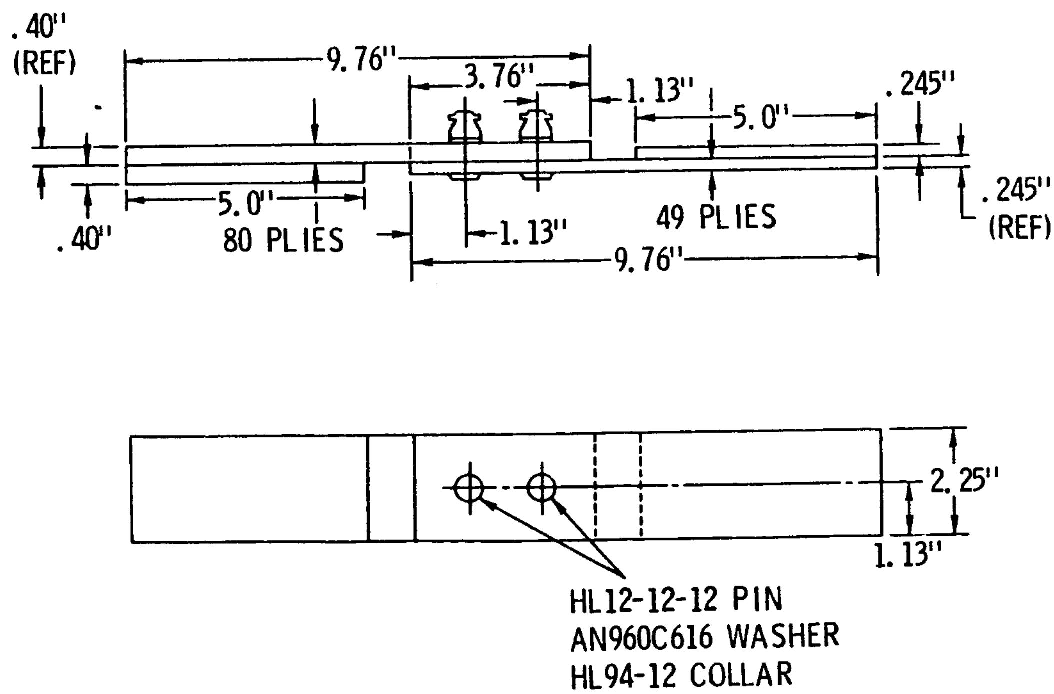


Figure 57. Concept Selection Joint Test Specimen - CS2-3

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6.8 CS3 SPECIMEN

The CS3 concept selection specimen is the design of an wing upper surface chordwise joint at the 55 percent semi-span of the 1993 LFC transport wing. This design concept will be developed and used as the end chordwise joint for both the concept verification and the concept demonstration panels. Figure 58 shows the CS3 concept selection specimen including definition of the overall specimen dimensions. The hat stiffener and surface skins are shown as they are transitioned into the joint area. Figure 59 shows the cross section of the CS3 concept selection specimen at the slot duct centerline. The LFC suction flow system is indicated by curved arrows with collection in the hat stiffener. Figure 60 shows the hat stiffener portion of the CS3 concept selection specimen. Cross-sections are taken at four locations along the hat stiffener with identification of the number of graphite/epoxy plies at each cross section.

6.9 CS4 SPECIMEN

The CS4 concept selection specimen test is the last in the series of the concept selection specimen group. The purpose of this test is to evaluate the LFC wing surface chordwise joint design at the front spar intersection including the spar cap splice. One-half of this joint specimen was fabricated of steel in the interest of minimizing costs. Figure 61 depicts the CS4 concept selection specimen and gives the overall specimen dimensions. The mechanical fastener patterns are shown as well as the stabilizing support plate.

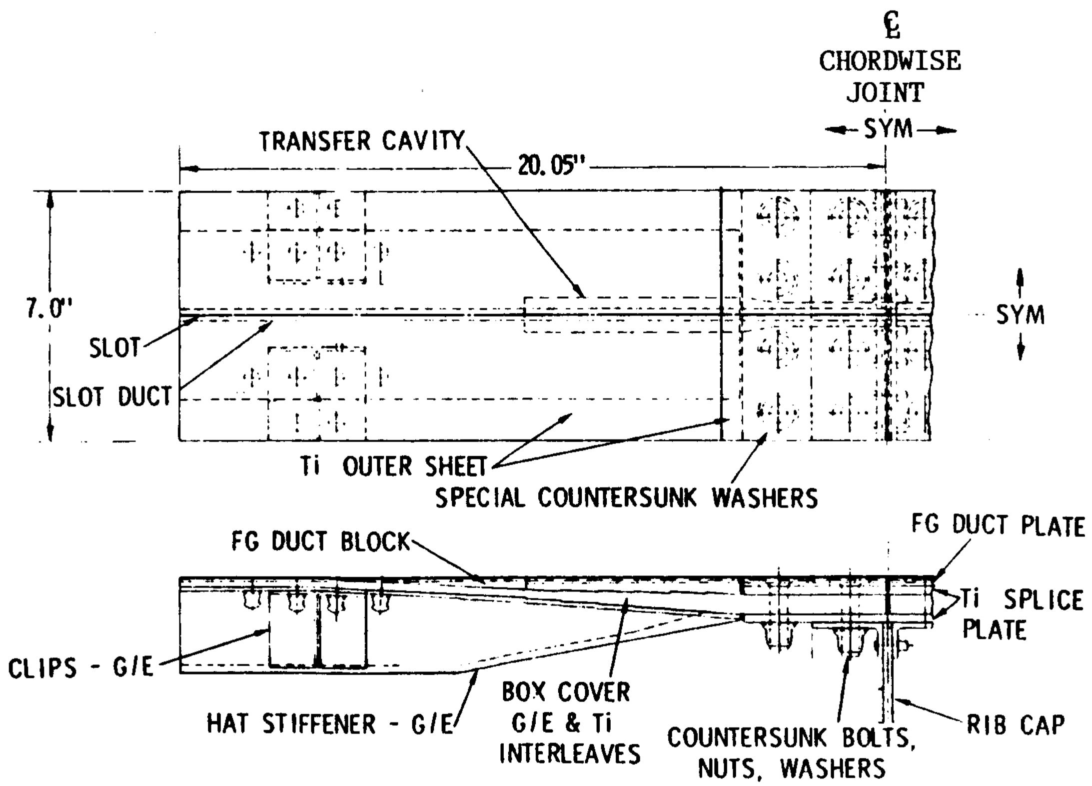

Figure 58. Concept Selection Specimen CS3

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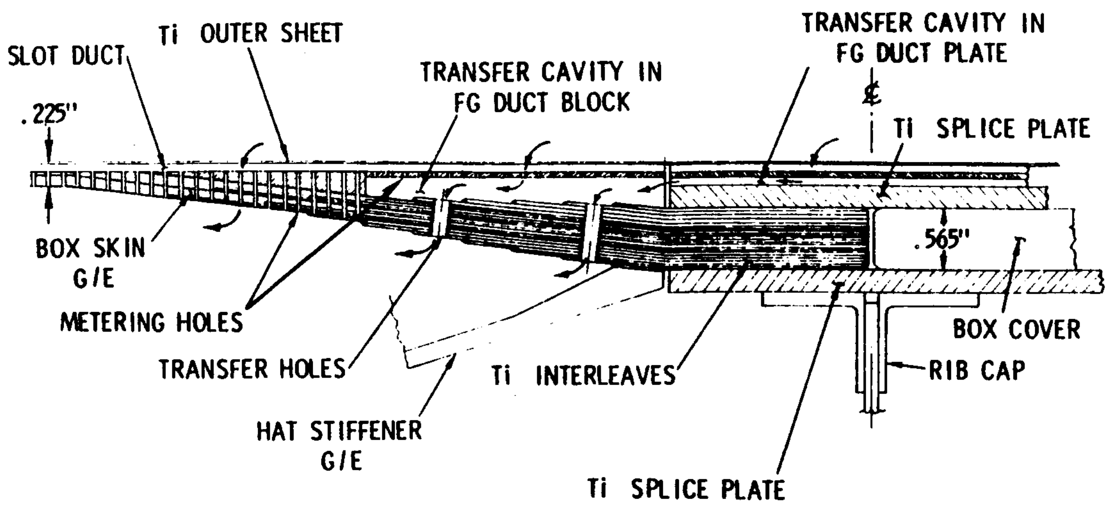

Figure 59. Concept Selection Specimen CS3 Cross Section at Slot Centerline

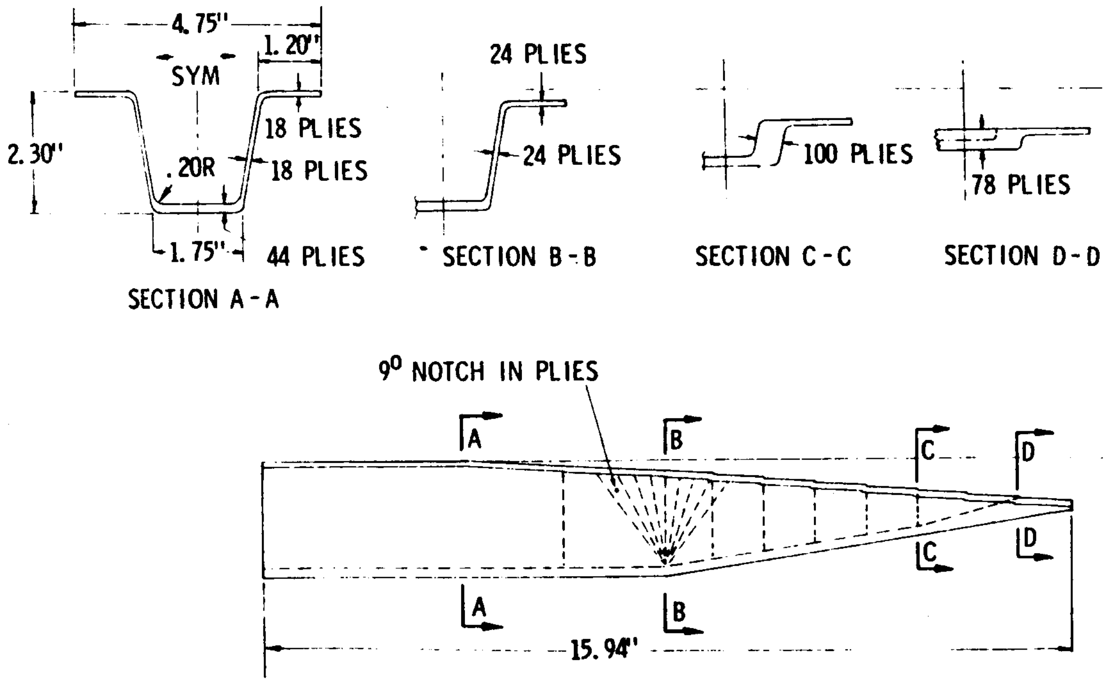

Figure 60. Concept Selection Specimen CS3 Hat Stiffener

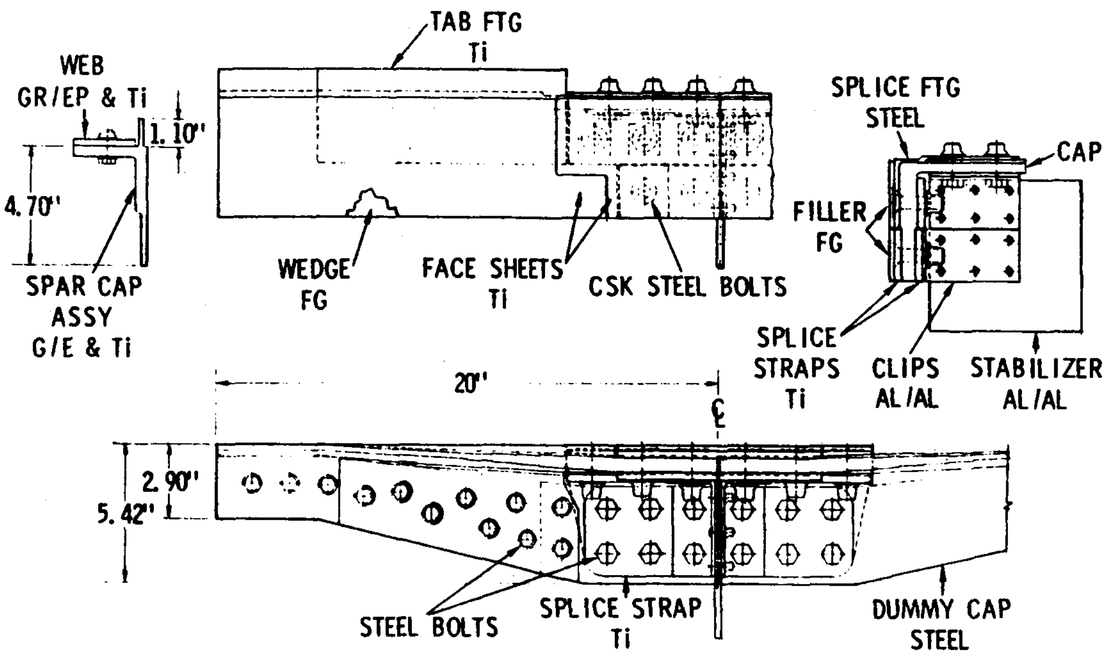

Figure 61. Concept Selection Specimen CS4

6.10 CONCEPT VERIFICATION SPECIMEN

A plan was prepared to verify the structural concepts and demonstrate the integrated LFC system in the wing surface of a future transport. A representative section of the wing surface was selected which is typical of the wing surface of a long-range transport. The area selected for the detail design, fabrication, development, and demonstration is located at the 55 percent semispan as shown in Figure 62.

Design surface loads at selected semispans are tabulated for the wing box of the long-range transport. Surface loads for the 55 percent semispan location, Table 8, were used for structural analyses.

General requirements and plans for the design and development of the concept verification panel are:

Design Loads:

- N_X = 15.9 kips/in
- N_{XY} = 1.7 kips/in

Panel Basic Dimensions

- Width = 79.68 in
- Length = 102.00 in
- Radius = 790.60 in

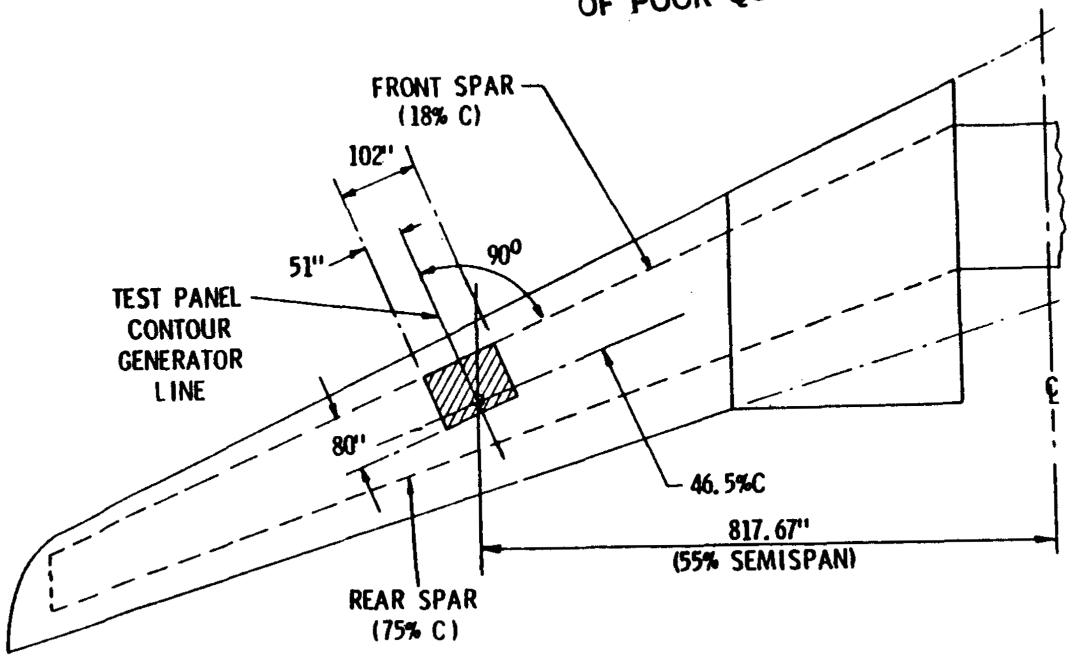

Figure 62. Test Panel Location

CV4 Tool Try Panel

- (2) CV1 Test Panels
- (3) CV2 Test Panels

CV4 Test Panel

- Install in Box-Shaped Section
- Test CV4 Panel/Box Assembly in C-130 Wing Test Fixture

A preliminary design of the concept verification panel was completed and shown in Figure 63. The panel has a chord of 80 in and a span of 102 in. It contains the chordwise splice, spar cap, two duct ribs, one typical rib and the LFC wing surface metering and ducting system.

The first concept verification panel fabricated on the new tool will be a tool try panel. This panel will be cut into the smaller specimens shown in Figure 64 for component testing. These smaller panels are identified as CV1 and CV2. The ancillary test plan calls for a third panel, CV3, for flow measurements under load. CV1 will be used for CV3 tests prior to the CV1 structural tests.

A cross section of the CV4 tool try panel is shown in Figure 65. This figure shows the location of the hats, slots and spar caps. The arrangement shown permits the CV1 and CV2 specimens to be cut out of the tool try panel.

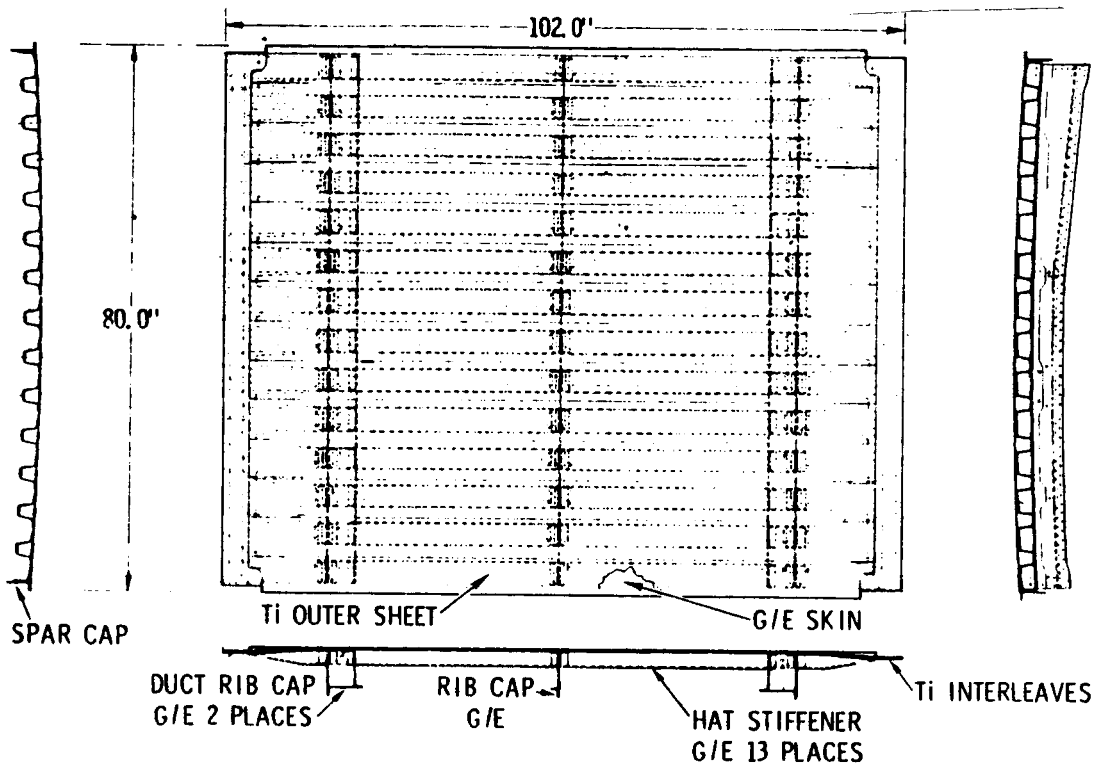

Figure 63. Test Panel Preliminary Design

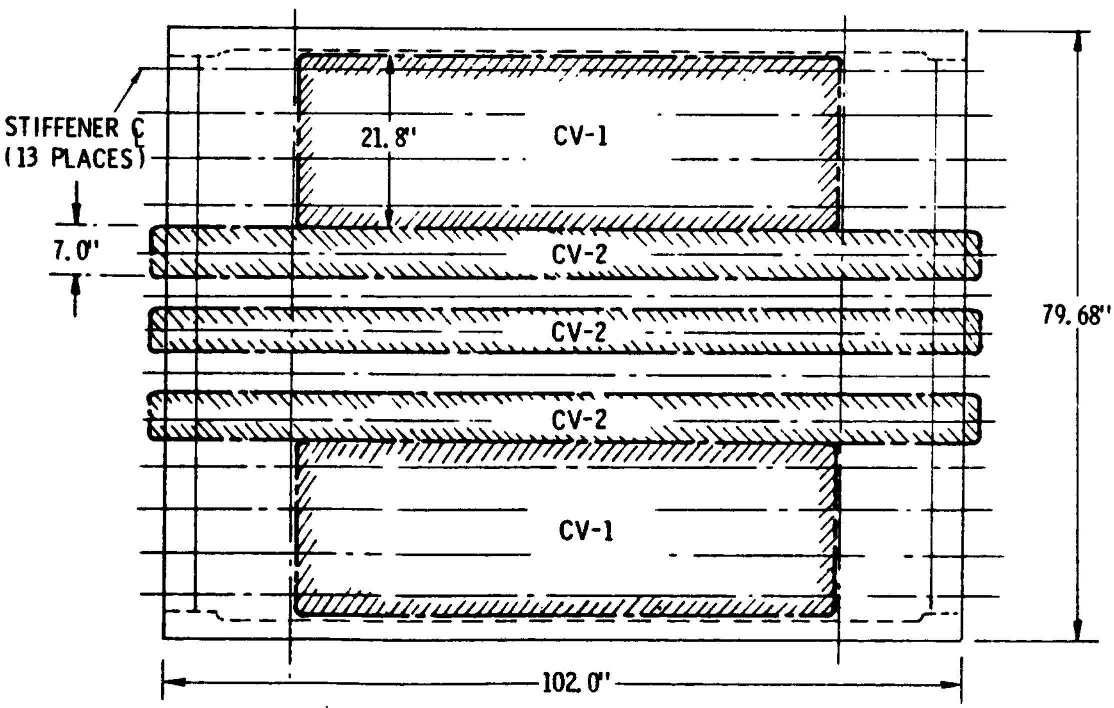

Figure 64. CV4 (Tool Try) Panel

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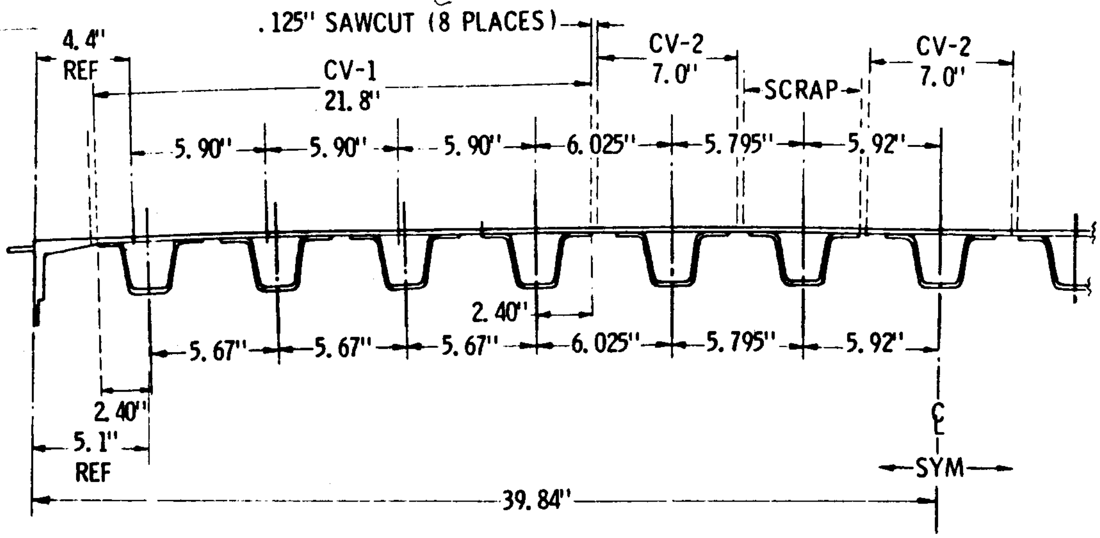


Figure 65. CV4 (Tool Try) Panel Cross Section

6.10.1 CV1 Specimen

Two CV1 test panels will be cut out of the tool try panel. The dimensions of these panels are:

- Four hat stiffeners per panel
- Hat stiffeners spacing = 5.167 in
- LFC slot spacing = 5.90 in
- Panel width = 21.8 in
- Panel length = 60.0 in

CV1 is a long column, compression specimen.

6.10.2 CV2 Specimen

Three CV2 test panels will be cut out of the tool try panel. Figure 66 shows the CV2 test specimen. Three specimens will be assembled as shown and used for tension, compression and fatigue testing.

6.10.3 CV4 Specimen

The CV4 test panel is shown in Figure 63. A test box simulating a wing box will be used for testing the concept verification panel CV4. The box will apply axial and torsional shear loads similar to those in an actual wing box.

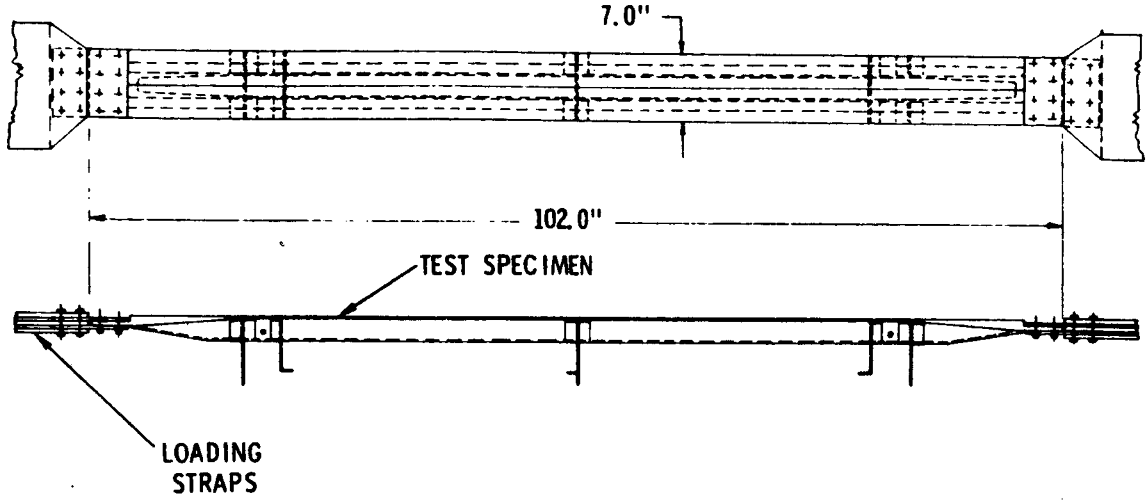


Figure 66. CV2 Test Specimen

The plan for modifying the C-130 center wing test fixture is shown in Figure 67. Although design and modification of this test fixture was not scheduled to begin until later, the current, proposed modification is shown in this illustration. Some of the features of this test fixture are:

- o Test box assembly is attached to dummy boxes mounted between two end loaders.
- o Bending moment is applied by four load actuators.
- o Torsion load is applied by four load actuators.
- o Closed-loop servo-controlled loading arrangement with load measurements from load transducers in series with load actuators.

Loads will be applied up to the limit design load. Surface smoothness, slot widths and flow measurements will be made at no load and under loading conditions.

6.11 CONCEPT DEMONSTRATION PANEL

The planned concept demonstration panel will incorporate the LFC suction system design refinements developed and verified in the ancillary test program. The concept demonstration panel is to be fabricated with the same tool used in fabricating the CV4 test panels. The demonstrations performed with this panel will establish the design and manufacturing feasibility of an LFC transport wing surface.

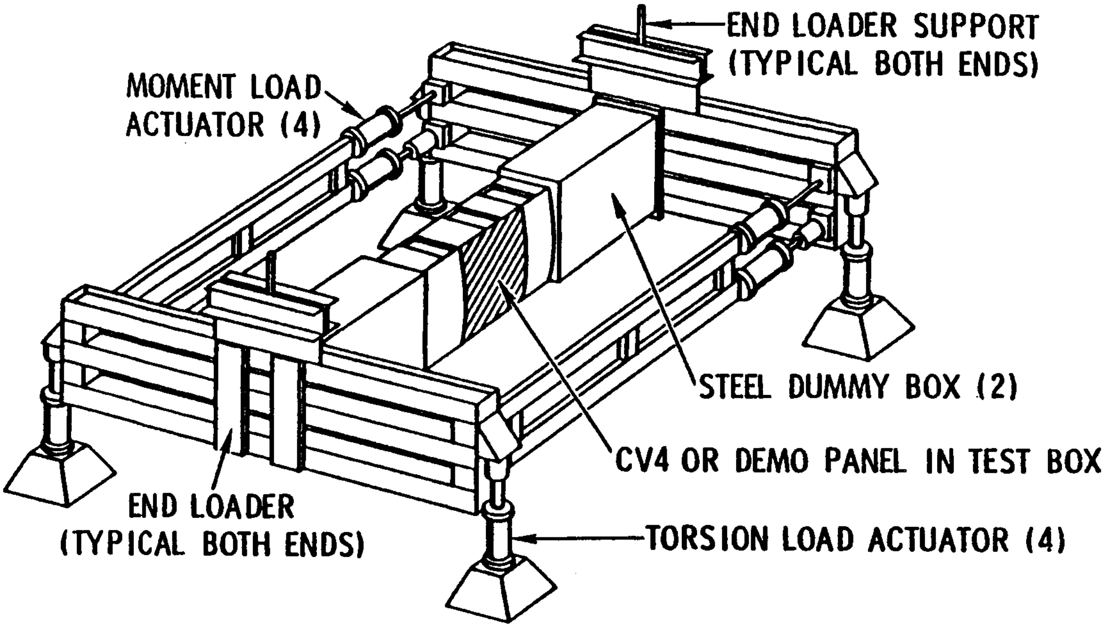


Figure 67. CV4 Panel Test Fixture

7.0 METHODS OF ANALYSIS

The aerodynamic configuration from the Phase I studies was used to provide the input data to the structural loads program, ANSWER, and to the flutter analysis program, ARF.

The wing torsional stiffness distribution required for flutter prevention was determined by means of the flutter optimization computer program ARF. This program operates in conjuction with the structural synthesis program ANSWER to develop minimum weight wing designs which satisfy both strength and flutter speed constraints.

Considerable resources have been expended at the Lockheed-Georgia Company for the development of computerized methods of analysis. Figure 68 lists the computer programs used extensively in this report.

Program LG031 predicts buckling loads for flat panels, such as spar webs, subjected to biaxial compression or combined tension and shear inplane loads. Program LG080 predicts peak stresses around cutouts in webs. Program LG041 calculates margins of safety for an integrally hat-stiffened surface panel subjected to combinations of inplane biaxial loads and shear plus normal pressure. In this program, a beam-column analysis is performed and margins of safety are calculated for several critical modes of failure. A partial list of modes of failure are:

- Maximum stress or strain in skin at rib or mid-bay between stiffeners
- Maximum stress or strain in stiffener at rib or mid-bay
- Initial buckling of skin
- Local buckling of stiffener crown or leg

Program LG014 predicts the load in each element of an open section given the total load and dimension of each element of the section. This program is used in the sizing of spar caps. Program LG062, used primarily in sizing rib struts, calculates margins-of-safety for open sections subjected to biaxial inplane loads and shear plus normal pressure. Margins-of-safety are calculated for several critical modes of failure, including:

- Strain in skin
- Strain in stiffener flanges or web
- Local buckling of stiffener flanges or web
- Torsional/flexural buckling of stiffener

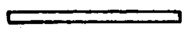
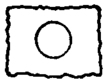
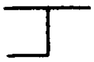
TYPE	CONFIGURATION	APPLICATION					LOADING CONDITION		
			SPAR		RIB				
		COVER	WEB	CAP	CAP	STRUT	BIAXIAL	SHEAR	NORMAL PRESSURE
FLAT PANEL			X				LG031	LG031	
FLAT PANEL CUT OUT			X				LG080	LG080	
SKIN/STRINGER	HAT 	X					LG041	LG041	LG041
	TEE 			X			LG014	LG014	
	JAY 		X		X	X	LG062	LG062	LG062

Figure 68. Computer Analysis Programs

8.0 MANUFACTURING PROCESSES AND PROCEDURES

Prior to WSSD contract award, the material used in Phase I, NARMCO T300/5208, was revised to the DV special formulation as a result of problems at Lockheed and General Dynamics in processing the 5208 system. The DV special formulation was a devolitalized version with a more precise advancement.

Panels, 12 in x 12 in, 40-ply laminates to the same orientation as planned for the surface panel skin, were laid up to check processing of thick laminates. A total of 10 laminates was made from Narmco T300/5208DV. None was perfect, but an acceptable process was developed.

Table 14 shows the extreme sensitivity of Narmco T300/5208DV to breathing during cure. A breather with holes punched on 0.5 in centers was required to vent the laminates. The Airtech A4000-P3 film had definite holes that allowed resin to flow as well as venting air. Trials with breathers with smaller holes which allowed only airflow were not successful. Prebled parts could be cured without such attention to venting. Therefore an open breather with an open overbleed must be used. The cure cycle as shown in Figure 69 for the T300/5208 DV material generally produced acceptable laminates.

After the early problems with the DV special formulation, Hercules AS4/3502 material was considered since General Dynamics was qualifying it to the same specifications as NARMCO T300/5208.

The first lot of Hercules AS4/3502 graphite/epoxy tape material was investigated in a process development study similar to that conducted with the Narmco T300/5208 tape material as presented in Figures 70 and 71. Four panels were fabricated with AS4/3502 material using different bleeds and breathers. All four panels were cured in a single autoclave run and all the panels were found to be acceptable.

The photomicrographs, Figure 72, show the differences in the cured T300/5208 and AS4/3502 graphite/epoxy laminate systems. Note the tendency of the T300/5208 material to form long, laminar voids, whereas the AS4/3502 contains more discrete and discontinuous voids.

The photomicrographs, Figure 73, are examples of excellent quality laminates of both T300/5208 and AS4/3502 materials. It is noted that the T300 fiber resulted in more precise ply layers.

The photograph shown in Figure 74 of a rejected MV5 skin panel laminate illustrates the sensitivity of the T300/5208 material to slight processing variations. The skin panel was cured on the same platen as was a hat-section stiffener of the same material. Hat stiffener tooling was made from a heavy aluminum plate which heated at a slower rate than the skin panel during the cure cycle. A temperature gradient of 10-20°F existed across the panel for as much as an hour until the temperature stabilized. A portion of the skin panel contained heavy porosity, whereas other sections were of acceptable quality.

Initial process studies with the Hercules AS4/3502 material resulted in excellent quality panels, but the first skin panel laminate produced for the MV2 specimens contained heavy porosity. An investigation was conducted to determine the cause of the heavy porosity. It was determined that the process study laminates had a 3.5°F per minute heat rise and the rate of heating was uniform. Less than ten minutes were required to heat the laminates from 250°F to 270°F. Pressure was applied to the process study laminates after a 60 minute dwell at 270°F. The skin panel laminate for the MV2 specimens also experienced a nominal 3°F per minute heat rise, but it required more than 45 minutes to heat the panel laminate from 250°F to 270°F. The panel laminate was nearly jelled before pressure was applied, which caused heavy porosity.

The revised cure cycle for the AS4/3502 material is presented in Figure 71. This cure cycle has also been adopted for the T300/5208. No processing problems have occurred since adopting this revised cure cycle.

The decision was made to use the AS4/3502 as the primary 350°F graphite/epoxy tape material. Currently, it is more readily available, less expensive, and less sensitive to venting than the T300/5208 material. The AS4/3502 tape material was purchased to the General Dynamic Specification FMS2023 with a revised areal weight of 150 g/m^2.

SPECIMEN NO.	UNCURED RC %	BLEEDERS NO.	BREATHER SQ IN/HOLE[2]	DWELL VACUUM TIME MIN	DWELL VACUUM TEMP °F	DWELL PRESSURE TIME MIN	DWELL PRESSURE TEMP °F	THICKNESS PER PLY IN	RESULTS
1	34.6	1[1]	36	65	270	-	-	.0057	VOIDS, DELAM.
2	34.6	4	16	82	270	-	-	.0053	VOIDS, POR.
3	36.1	4	16	30 45	270 220	30	270	.0055	VOIDS, POR.
4	36.1	4	16	45 90	270 220	-	-	.0054	CRACKS, DELAM., VOIDS
5	36.1	4	16	30	270	60	270	.0055	SMALL DELAM., VOIDS
6	36.1	6	0.25	75	270	90	270	.0053	GOOD, SMALL POR.
7	33.1	4	16	60	270	90	270	.0054	VOIDS, DELAM.
8	33.1	6	0.25	75	270	90	270	.0053	GOOD
9	33.1	6	16	75	270	90	270	.0053	FAIR, SMALL POR. & DELAM.

[1] AIRTECH.
[2] MYLAR - PANEL 1, AIRTECH A4000-P3 TEFLON - PANELS 6 AND 8.

TABLE 14.

TABLE 14. SUMMARY OF NARMCO T300/5208 DEVELOPMENT SPECIMENS

- OPEN BREATHER (VENT ON .5 INCH CENTERS) MUST BE USED
- OPEN OVERBLEED REQUIRED
- CURE -

  HEAT TO 270°F AT 3°F/MIN UNDER VACUUM

  DWELL AT 270°F FOR 75 MINUTES UNDER VACUUM

  APPLY 85 PSI PRESSURE

  DWELL AT 270°F, 85 PSI, 29" VACUUM FOR 90 MINUTES

  HEAT TO 350°F

  CURE AT 350°F, 85 PSI, FOR 120 MINUTES

  COOL TO 160°F UNDER PRESSURE
- PREBLEEDING DURING LAYUP REDUCES NEED FOR OPEN VENT

Figure 69. Processing Summary - NARMCO T300/5208 DV

NO. OF SPECIMENS

4 12 IN. x 12 IN. 40 PLY LAMINATES

BLEEDER/BREATHER STUDY

NO. 1 - 6 PLIES OF BLEEDER S VENTED ON 4 IN. CENTERS

NO. 2 - 6 PLIES OF BLEEDER S VENTED ON 0.5 IN. CENTERS

NO. 3 - PREBLED - 1 PLY OF BLEEDER S VENTED ON 4 IN. CENTERS

NO. 4 - PREBLED - 1 PLY OF BLEEDER S VENTED ON 12 IN. CENTERS

CURE

USED T300/5208 SCHEDULE

RESULTS

ALL PANELS WERE EXCELLENT

Figure 70. Development of Processing for Hercules AS4/3502

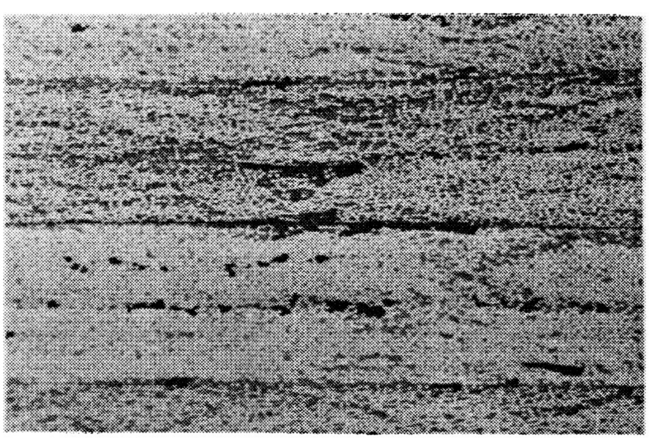
REVISED CURE CYCLE

- APPLY VACUUM - HEAT TO 270°F AT 3°F/MINUTE
- RECORD TIME AT 250°F
- DWELL 30 MINUTES AT 270°F UNDER VACUUM
- DO NOT EXCEED 50 MINUTES AFTER 250°F WITHOUT FULL PRESSURE
- APPLY 85 PSI
- DWELL AT 270°F, 85 PSI, FULL VACUUM FOR 90 MINUTES
- RELEASE VACUUM AND HEAT TO 350°F AT 3°F/MINUTE
- CURE 120 MINUTES AT 350°F UNDER 85 PSI
- COOL TO 160°F UNDER PRESSURE

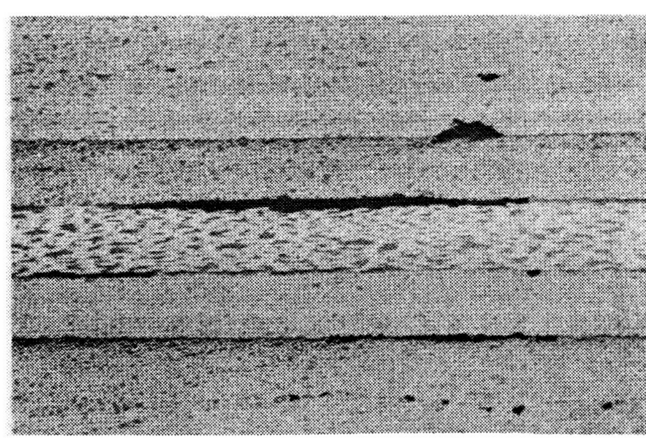
RESULT

- GOOD LAMINATE
- REVISED CURE CYCLE APPLIED TO T300/5208 WITH GOOD RESULTS

Figure 71. Development of Processing for Hercules AS4/3502



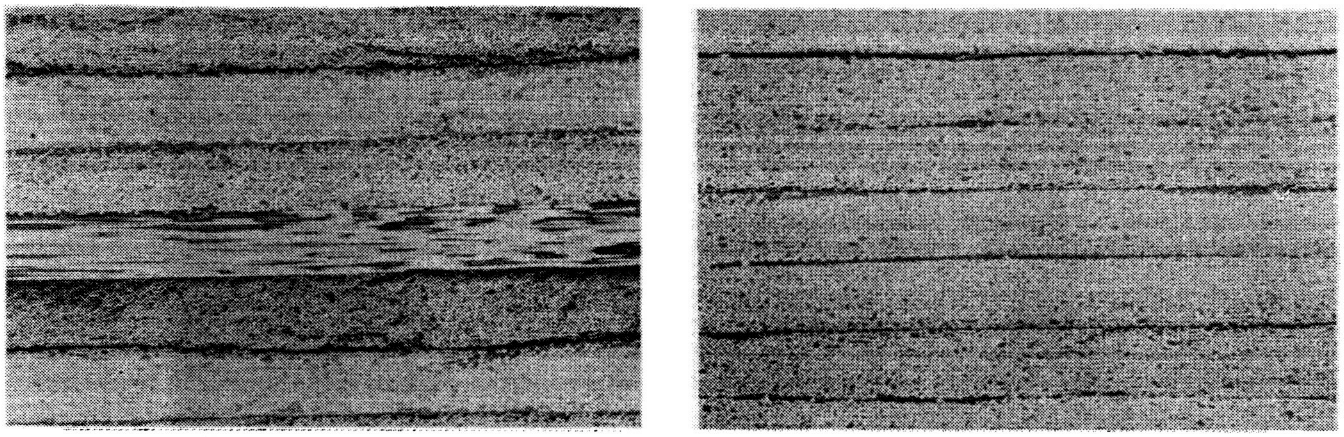
HERCULES AS4/3502
(MAG 100X)



NARMCO T300/5208
(MAG 100X)

HIGH POROSITY

Figure 72. Comparison of Porosity in Thick Laminates



HERCULES AS4/3502
(MAG 100X)

NARMCO T300/5208
(MAG 100X)

VOID FREE

Figure 73. Comparison of Porosity in Thick Laminates

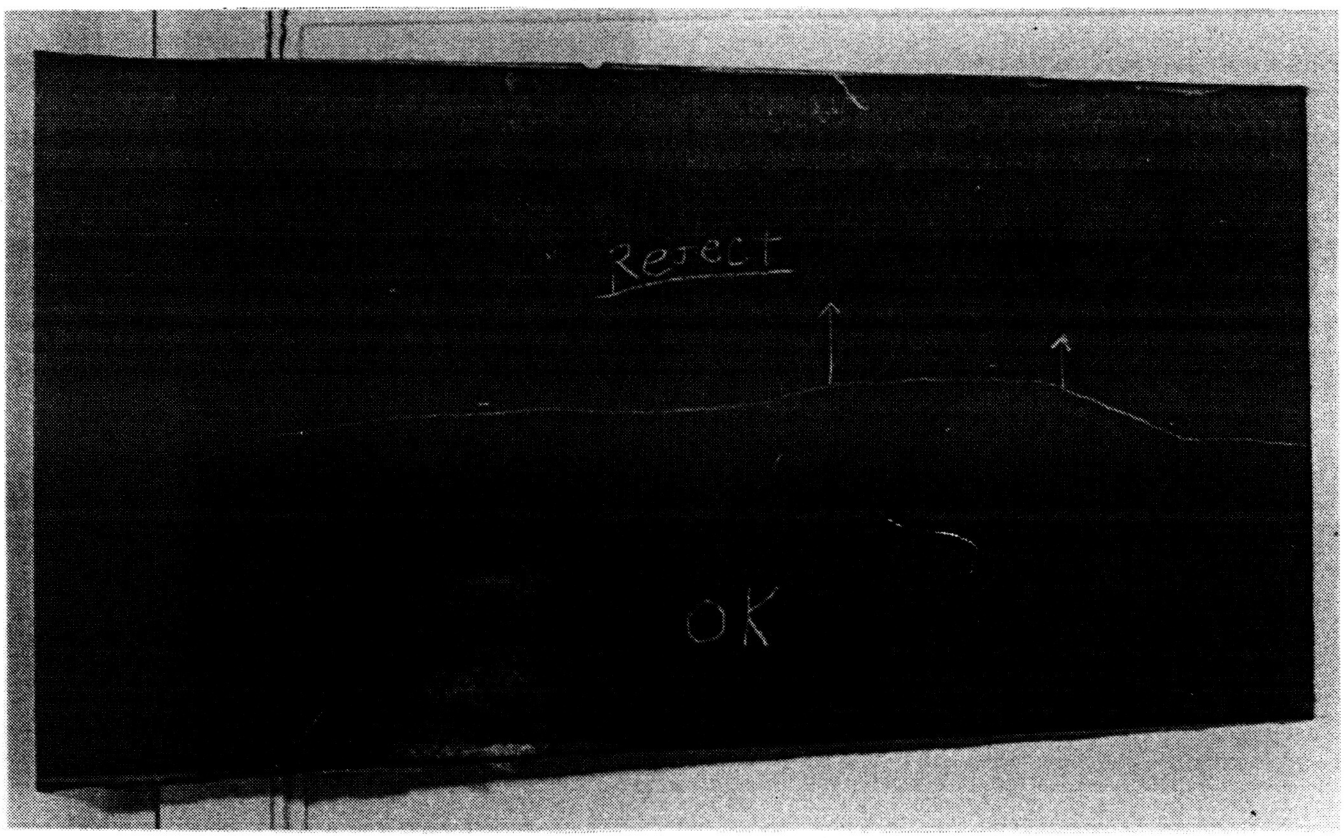


Figure 74. Effect of Temperature Gradient Across Platen

9.0 TOOLING

One of the major accomplishments of the WSSD project was the development of tooling concepts for thick composite structure. Hard tooling to the outer surface contour was necessary to hold the rigid surface contours and the strict requirements for surface smoothness. Skins with 40 to 50 plies of graphite/epoxy tape were fabricated on the surface tools with low resin content material. Hat stiffeners and spar caps with thicknesses exceeding 0.5 in were successfully fabricated free of voids and porosity with the tools developed for these components. Complex tooling for transition of the hat stiffened surfaces and the spar caps into chordwise joints were made and verified during the WSSD project.

Skins were cured on a flat aluminum plate with a shaped aluminum strip tack riveted to the plate to form the slot duct as shown schematically in Figure 75. The first sixteen plies were laid up and cut into strips to fall between the slot duct formers. Mold release was used over the entire tool to allow removal of the part from the tool after curing. Figure 76 shows the 12 in wide prepreg tape used to fabricate the skins. Also shown is the skin tool with the first 16 plies in place.

Sealing of the layup to control resin bleed was found to be extremely important. Narmco 5208 and Hercules 3502 both have an extremely deep viscosity drop of initial heating, reaching one to two poise at 270°F. Sealing the tool and preventing free resin flow are vital if void free laminates are to be produced.

Hat stiffeners were laid up in a female aluminum tool shown schematically in Figure 77. Great care was required to assure that the prepreg nested into the inside radius. In fact the radius had to be increased. Also, care in layup and bagging was necessary to form a good outside radius at the flange. Sealing was found to be even more critical on the hat stiffeners than on the skins since it was inherently more difficult to seal these tools. Further development in later tool designs paid more attention to designing the tools for sealing. The inner rubber plug was found to be necessary to form both the inside shape and the flange radius. Figure 78 shows the hat stiffener layup in progress. The right photograph shows the attempt to seal the tool ends to control resin flow. Later designs used a closed end mold to simplify resin sealing.

Tooling to produce the CS1 specimen was difficult and cumbersome. All details possible were precured. However, it would have been impossible to precure the fabric clip details as the design was based on these being laid up as a preform and cocured in place. Rubber blocks were used to provide pressure from expansion and from externally applied autoclave pressure. All prepreg details were laid up on the rubber blocks and positioned into place against the skin/hat/rib web details. It is much easier to lay up a highly formed item like the clips on a male rubber layup block than a female cavity. Difficulty was experienced in holding the precured frame assembly normal to the skin surface. It was finally necessary to use an external fixture to prevent movement under autoclave pressure and temperature. Figure 79 shows the many precured details, uncured details, and tooling parts.

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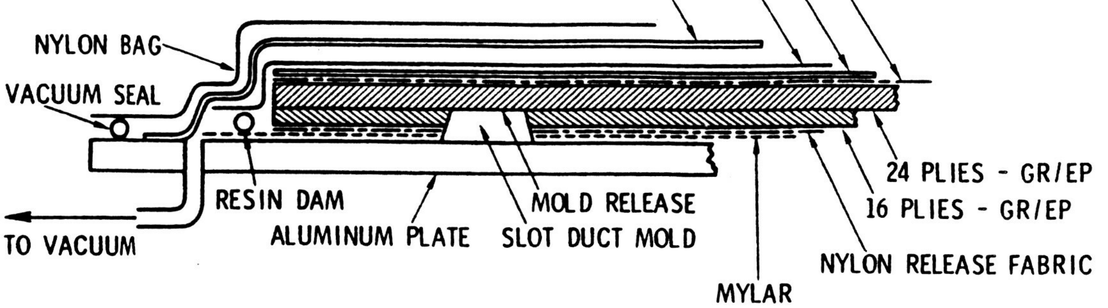
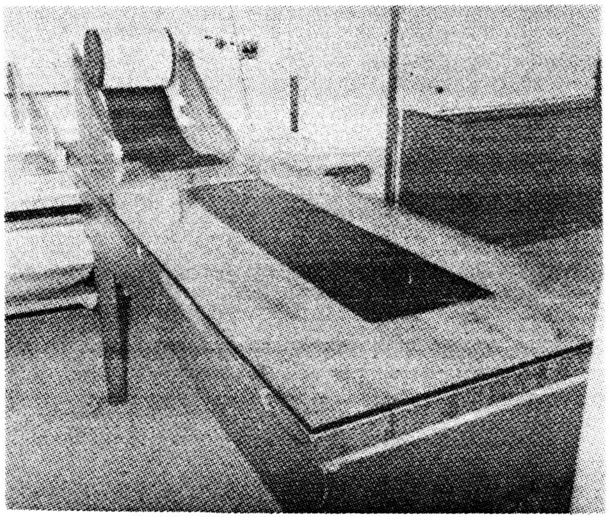
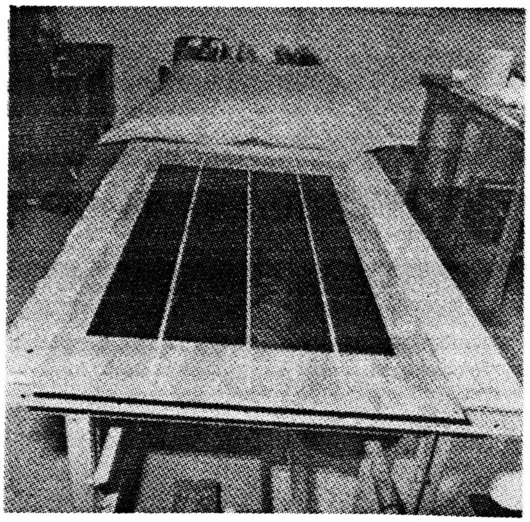


Figure 75. Schematic of Skin Layup for Cure



NARMCO T300/5208 TAPE



SKIN PANEL LAYUP ON TOOL
FIRST 16 PLIES DOWN

Figure 76. Graphite/Epoxy Skin Layup

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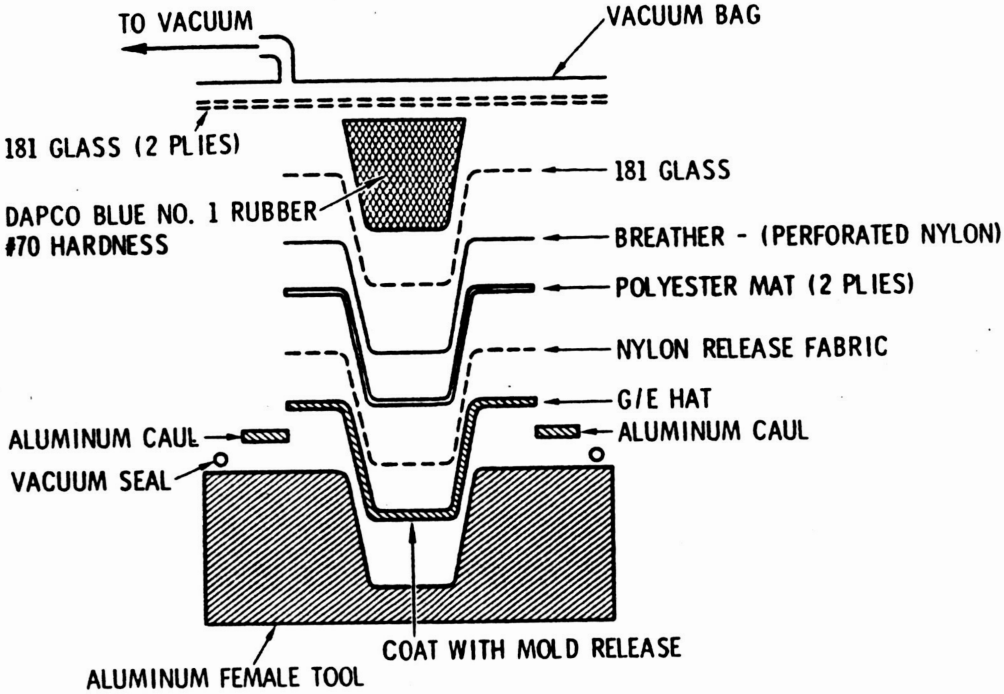
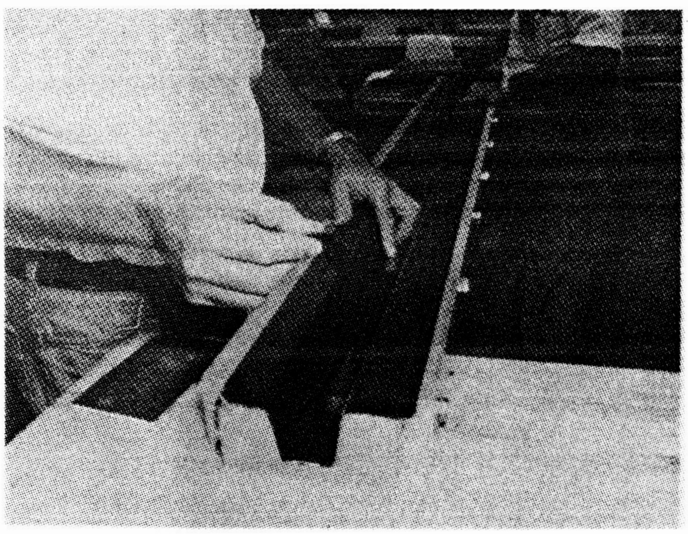
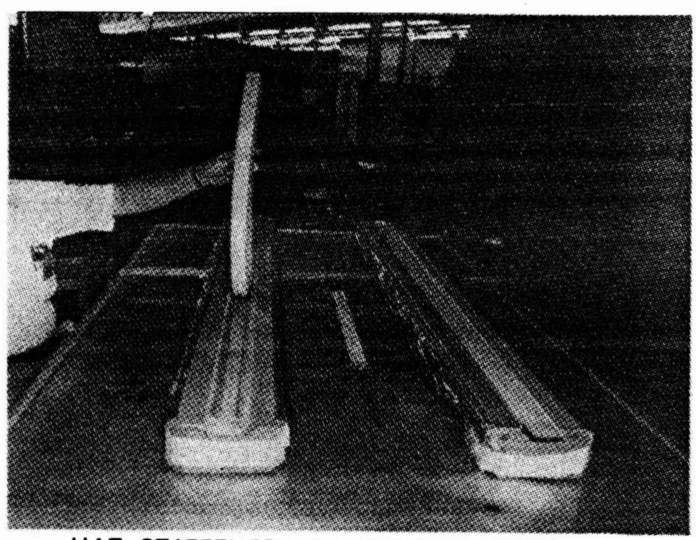


Figure 77. Schematic of Hat Layup for Cure

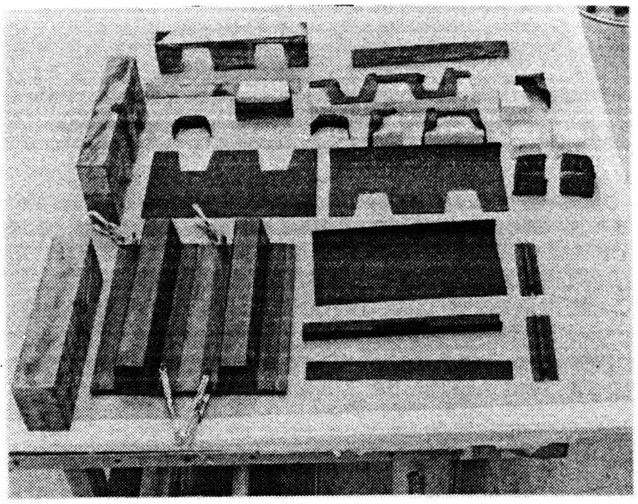


HAT STIFFENER TOOL WITH
LAYUP IN PROGRESS

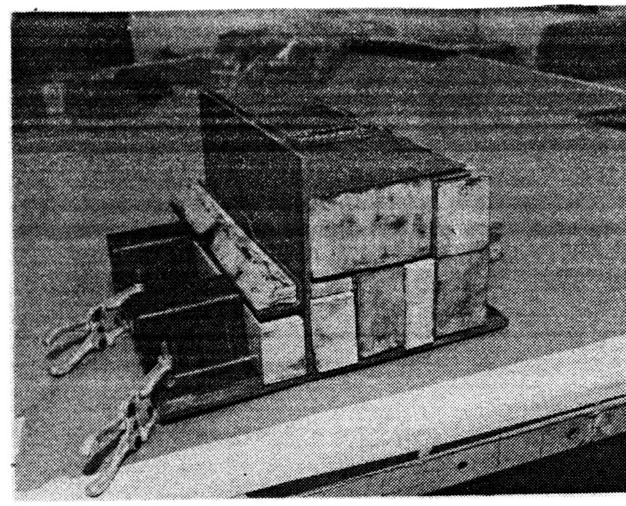


HAT STIFFENERS READY FOR CURE

Figure 78. Fabrication of Hat Stiffener



DETAIL PARTS & TOOLS



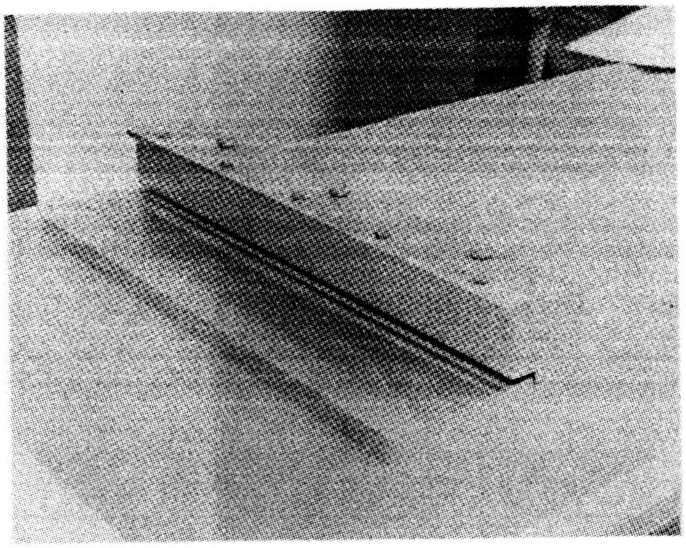
ASSEMBLY READY FOR CURE

Figure 79. Assembly of CS1 Specimen

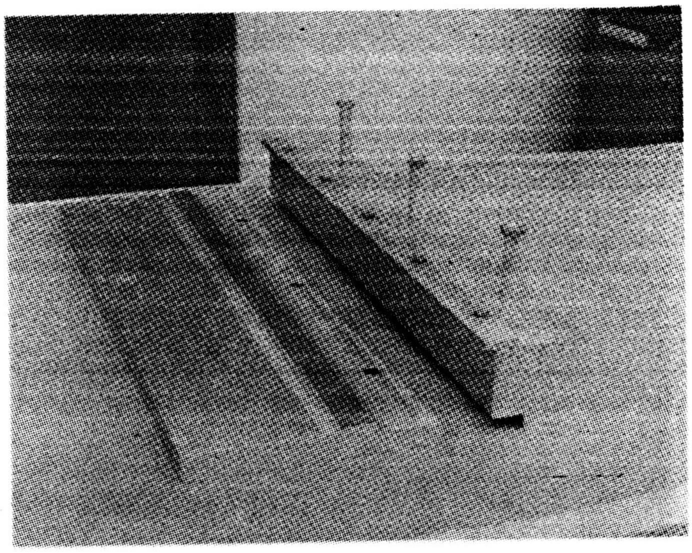
Fabrication tooling for the CS2 spar cap specimen was based on experience gained on prior specimens. It would have been impossible to bleed a laminate this thick, so it was planned to tool for no bleed. Hard precision tooling was needed to form the detail shape required for a spar cap. Figure 80 shows the machined aluminum tool. The removal member was allowed to float downward to stops, but no lateral movement was permitted. It was thought that the bag would form the interior surface without tooling; however, this was a bad assumption. Figure 81 shows the result. Significant fiber wash occurred. A rubber angle block was then cast to the desired internal dimensions of the spar cap. The second specimen was fabricated using this inner surface tool, which completely encapsulated the layup except for the ends, which were sealed with "Air Dam" putty. A perfect part was produced. Sections of the spar cap specimen are shown in the left photograph of Figure 81. Note the improvement in the cross-section between the first and second test specimen due to the use of the rubber filler blocks shown in the right photograph of Figure 81.

Ply templates required to lay up the CS3 hat stiffener are shown in Figure 82. Tooling shown in Figure 83 was developed for fabrication of the CS3 chordwise joint. The left photograph shows the hat-section stiffener mount tool and the right photograph shows the center rubber mold block for the hat-section stiffener.

The CS4 specimen was the most complex specimen produced under the contract. Accordingly, the tool was the most complex and precise to be designed. Again, the accumulated experience from all previous specimens was used to design and fabricate the tools for CS4. More attention was given to sealing the tool to prevent resin flow; therefore, an excellent part was produced on the first run with no problem.

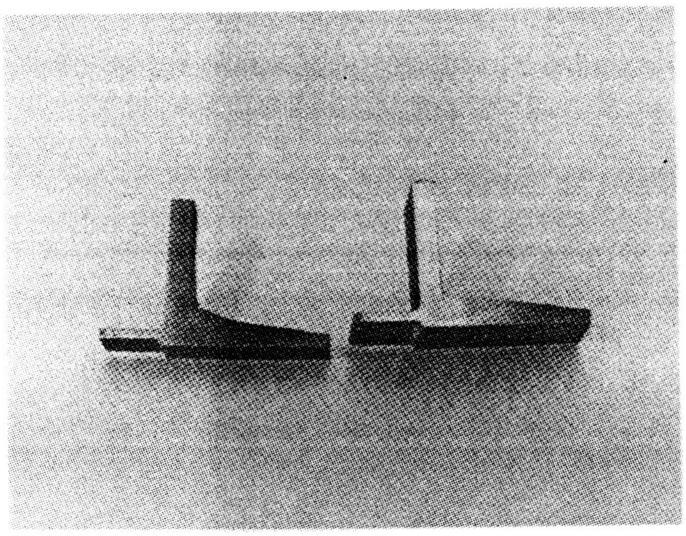


ASSEMBLED

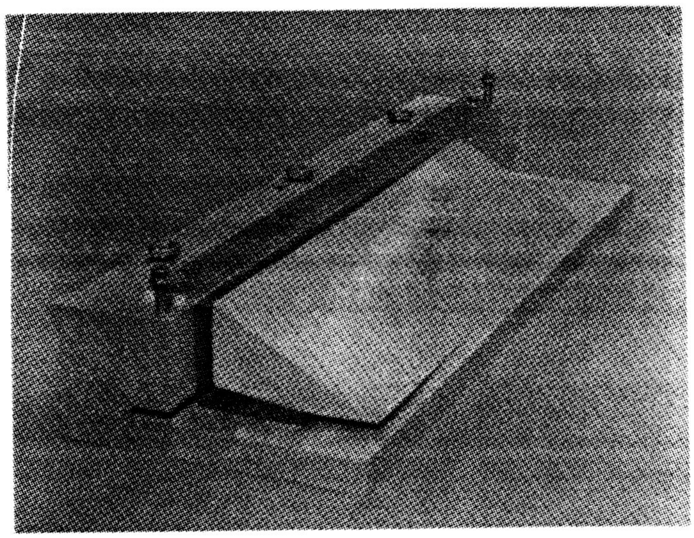


DISASSEMBLED

Figure 80. CS2 Spar Cap Specimen Tool



SECTION OF 1ST AND 2ND
SPAR CAPS



RUBBER TOOL BLOCK
USED TO ELIMINATE
FIBER WASH IN
2ND SPECIMEN

Figure 81. CS2 Spar Cap Specimen and Tool

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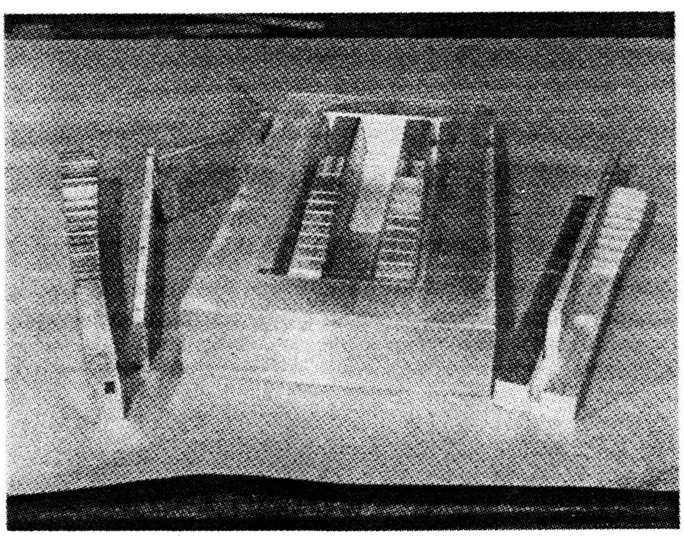
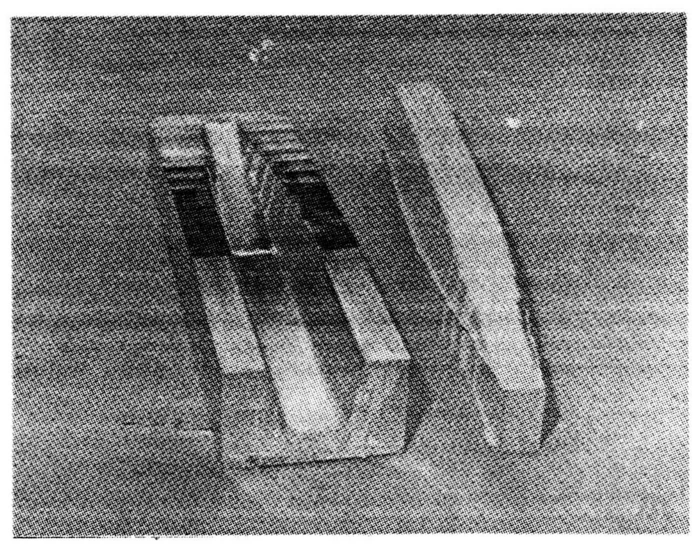
Figure 82. Ply Templates



HAT RUNOUT



CENTER RUBBER
AND MOLD

Figure 83. CS3 Chordwise Joint Tools

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10.0 QUALITY ASSURANCE

The purpose of the development of surface acceptance criteria shown in Section 5.7 is to establish a baseline for identifying inspection methods and measurement equipment for evaluation of the 1993 LFC aircraft wing structure. A preliminary assessment of inspection methods and measurement equipment indicates that commercially available equipment, Table 15, is adequate except for surface waviness and surface step measurements. Surface waviness and surface steps are planned to be measured by a Lockheed designed ripple measurement unit (RMU).

Quality of flat, composite skins was evaluated using ultrasonic "C" scans, measured thicknesses, and resin contents. Quality of corners in hats and other shapes were based on destruct tests of first articles and proving the tool.

Adhesive bonding was used extensively for assembly of structural elements. Fit up of mating surfaces and normal processing of the adhesives and adherends provided excellent bonds. After prefitting the bondlines were checked using verifilm. Where accessible, bonded parts were checked using ultrasonic "C" scans. Blind areas depended on control of the surface treatment, fit up and control of the bonding process. Problems were found in the control of the surface treatment of the rib clips to rib cap and a weak bond resulted. This problem was corrected by careful control of the surfaces prior to bonding.

CATEGORY/NOMENCLATURE	TYPE	PART NO. (OR EQUIV.)	SOURCE	APPLICATION
PROFILOMETER	COMM'L	SURTRONIC 3	TAYLOR-HOBSON	MEASURE EXTERIOR SURFACE ROUGHNESS
- PARAMETER MODULE	COMM'L	(TBD)*	TAYLOR-HOBSON	MODIFIES PROFILOMETER TO MEASURE MAXIMUM RANGE OF ROUGHNESS AMPLITUDE
DIAL INDICATOR	COMM'L	710J400	LOCKHEED	MEASURE STEP AT EXTERIOR SURFACE JOINTS AND SLOT EDGE
THICKNESS GAUGE	COMM'L	730G250	LOCKHEED	MEASURE GAP AT EXTERIOR SURFACE SLOTS
MAGNIFIER, MEASURING	COMM'L	730W005	LOCKHEED	MEASURE GAP AT EXTERIOR SURFACE JOINTS
RIPPLE MEASURE UNIT (RMU)	NEW	(TBD)	LOCKHEED	MEASURE EXTERIOR SURFACE WAVINESS
ULTRASONIC UNIT	COMM'L	MARK I OR IV	SONIC	MEASURE DELAMINATION/DISBOND OF OUTER SHEET BOND AREA
- TRANSDUCER	COMM'L	57A2214	AUTOMATION INDUST.	SENSES ULTRASONIC SIGNAL FOR 1/4-INCH DIAMETER DEFECT
- STANDARD, NDI	NEW	(TBD)	LOCKHEED	IDENTIFIES PROFILE OF ACCEPTABLE ULTRASONIC READING
SUCTION TEST SET	NEW	(TBD)	LOCKHEED	TO CHECK-OUT SUCTION FLOW RATE IN SLOT-DUCT SYSTEM

* TO BE DETERMINED

TABLE 15.
OPERATIONAL INSPECTION SUPPORT EQUIPMENT
LFC WING PANEL

C-2

11.0 FABRICATION AND ASSEMBLY OF TEST SPECIMEN

11.1 FABRICATION OF GRAPHITE/EPOXY SKIN

The schematic of the graphite/epoxy skin layup is shown in Figure 75. The aluminum slot duct molds were tack riveted to the aluminum tool plate. Mylar and nylon release fabric were next placed on the slot duct mold and aluminum plate. The graphite/epoxy skin was laid up in three modules. Module number 1 was made up of 16 plies and cut into strips to fit between slot duct molds, Figure 78. Module numbers 2 and 3 were made of 12 plies each laid on top of Module No. 1. The graphite/epoxy skin was prepared for curing by adding:

- o Nylon release fabric
- o Bleeder (polyester mat)
- o Breather (perforated nylon)
- o Glass cloth overbleed
- o Resin dam
- o Vacuum seal
- o Nylon vacuum bag
- o Vacuum tube

The left photograph in Figure 84 shows a skin panel laminate lay-up prior to application of release fabric and bleeders. The right photograph on the shows a small drill press being used to drill metering holes. These holes were drilled using #55 solid carbide circuit board drills. Drill speed was 8,000 to 10,000 revolutions per minute.

11.2 FABRICATION OF GRAPHITE/EPOXY HAT STIFFENERS

Figure 77 shows a schematic of the hat-stiffener layup for cure, with a female aluminum mold and a DAPCO Blue No. 1 rubber plug material used for tooling. The release coat, release fabric, and bleeder system are identified in the figure. The graphite/epoxy hat has 44 plies in the crown and 18 plies in the legs.

In Figure 78, the left photograph shows the hat-section stiffener tool with lay-up of a graphite/epoxy stiffener in progress. The hat-stiffener tool is a female aluminum metal tool contoured to the outside contour of the hat-section stiffener. The right photograph shows a bagged hat-section-stiffener/tool-assembly ready for autoclave cure.

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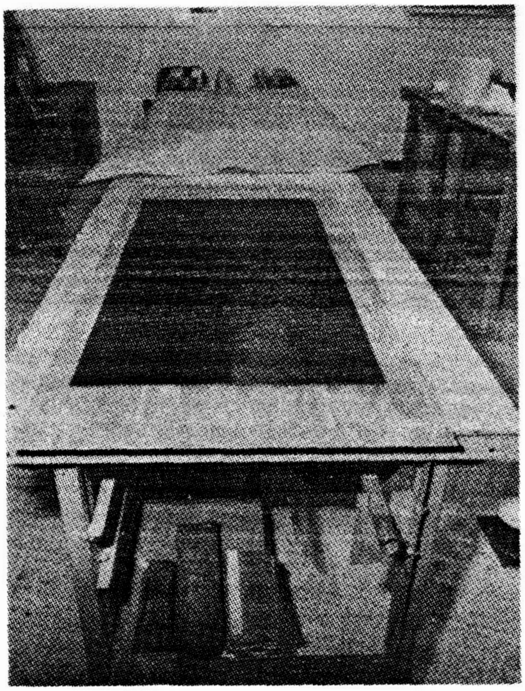
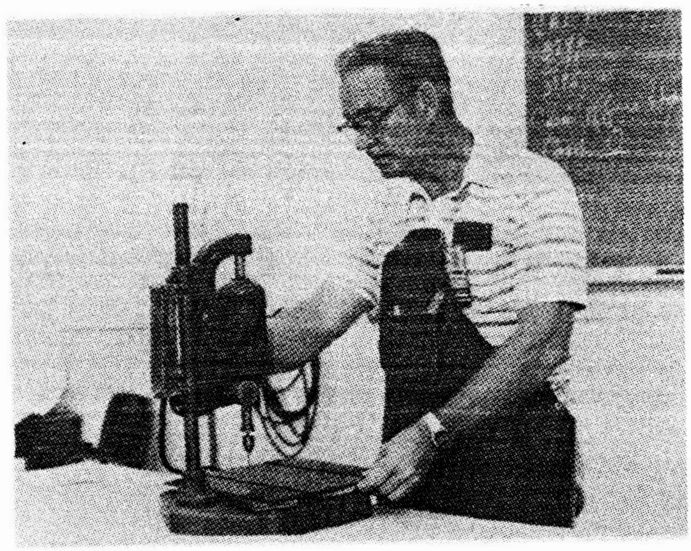


SKIN PANEL READY
FOR BAGGING



DRILLING METERING
HOLES IN SKIN PANEL

Figure 84. Graphite/Epoxy Skin Panel

11.3 ASSEMBLY OF TITANIUM SHEET, G/E SKINS AND G/E HATS

All bonding surfaces were prepared for bonding by abrasive cleaning with aluminum oxide grit followed by washing with a solvent. The hats were bonded to the graphite/epoxy skin using FM 73 adhesive cured for one hour at 250^{o}F and 35 PSI.

Cleaned and primed titanium sheet was then bonded to the prepared surface of the laminate with FM123-4 adhesive. The adhesive was cured at 200^{o}F at 30 PSI. After bonding, slots were cut in the titanium sheet using a high speed steel jeweler's saw. The titanium sheet graphite/epoxy bondline was then ultrasonically inspected. Figure 85 shows the completed MV5 specimen.

11.4 FABRICATION AND ASSEMBLY OF RIB-TO-SURFACE

The CS1 specimen was used to develop manufacturing procedures for assembly of the rib duct to the hats and wing surface. Figure 54 is a three-view presentation of the configuration of the CS1 concept selection specimen. During fabrication and test of the first CS1 specimen, difficulty was experienced in obtaining a good reliable bond. The tooling was revised as shown in the left photograph in Figure 86. The major steps in the revised fabrication plan for the CS1 specimen were as follows:

- o Precure details (hats, skin, cover, and angles)
- o Bond hats to skin and inspect

- o Cocure rib web to surface panel and inspect
- o Cocure duct wall to surface panel and inspect
- o Install rib clip fasteners
- o Install duct cover
- o Bond titanium and skin and slot

The right photograph in Figure 81 shows duct cover in place ready for fasteners.

11.5 FABRICATION OF SPAR CAP

Figure 56 shows the CS2 concept selection specimen which is the front spar cap of the 1993 LFC transport wing. The cross section of the specimen shows the variation in graphite/epoxy ply orientations, with the light area representing the $\pm 45^{\circ}$ plies and the dark area representing the 0° plies. The major steps in the fabrication plan for the first CS2 spar cap specimen were as follows:

- o Use graphite/epoxy - Hercules AS4/3502
- o Use hand tool on mating surfaces suitable for the Demonstration Panel
- o Precompact under pressure
- o Use vacuum bag to form inner corner

Inspection of the first specimen showed washing of fibers to the inner corner of the vacuum bag side of the specimen accompanied by some porosity.

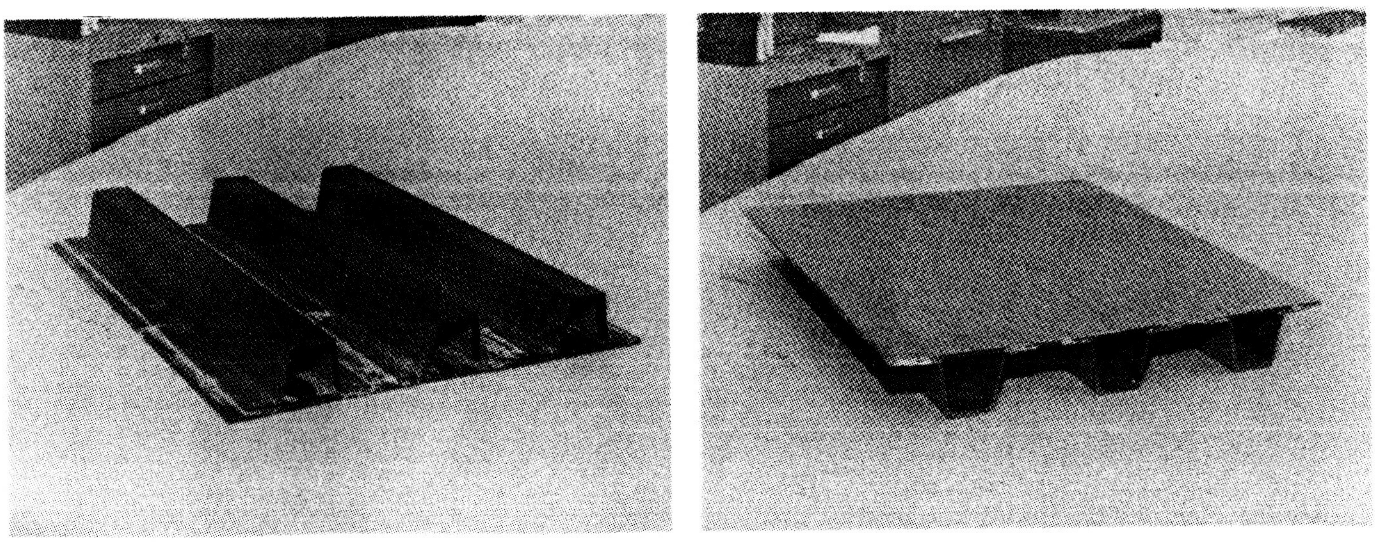
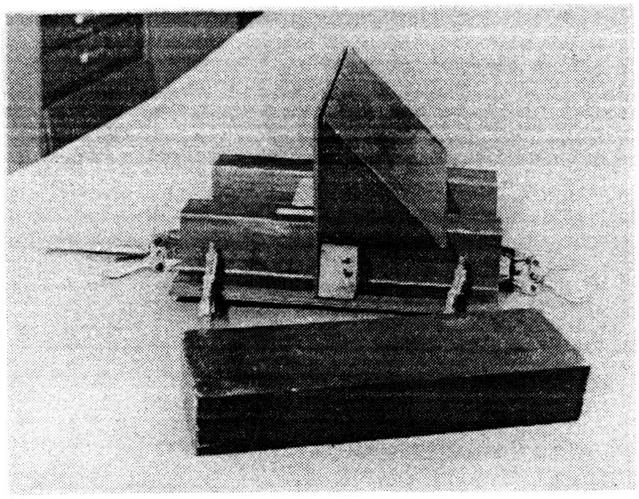
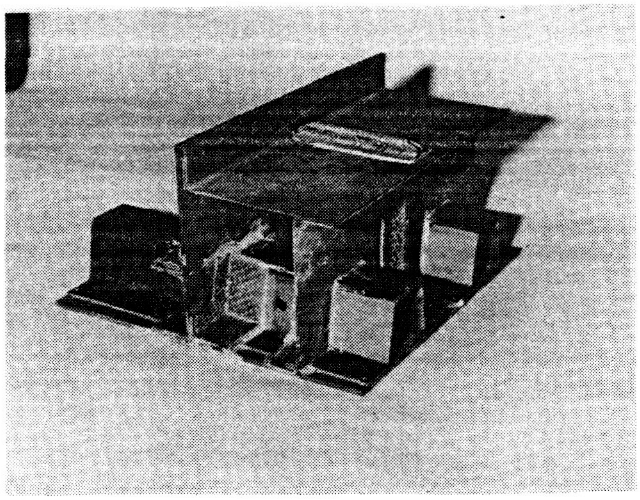


Figure 85. MV5 Specimen Ready for Potting Ends



REVISED TOOLING FOR
ASSEMBLY OF RIB DUCT



DUCT COVER IN PLACE
READY FOR FASTENERS

Figure 86. Development of CS1 Rib Duct Specimen

In fabrication of the second CS2 spar cap specimen, a rubber block was used to prevent fiber wash. The left photograph in Figure 81 shows cross-sections of both the first and second CS2 spar cap specimens. The right photograph shows the rubber tool block used to eliminate the fiber wash in the second CS2 spar cap specimen.

11.6 FABRICATION OF CHORDWISE JOINT

The third concept selection specimen, CS3, was used to develop the critical chordwise splice required in a typical wing surface. Figure 58 shows the CS3 test specimen complete with all associated hardware. The specimen length is that required for panel transition into the splice area plus 4 inches on each end to provide for installation of the specimen in the test fixture. The specimen width is the minimum necessary to achieve a satisfactory test of the splicing concept. Even though the stiffener spacing is 6 inches, the specimen was widened to 7 inches and the fastener spacing closed to approximately 3.75 fastener diameters to allow two rows of fasteners on each side of the stiffener centerline. The cover load at 55 percent semispan location is 15980 lb/in.

Figure 59 shows a cut through the chordwise joint. It shows all elements of the splice and transfer ducts across the joint. A slot in the titanium outer sheet, a slot duct in a fiberglas duct plate, and metering holes provide continuous LFC suction slot across the joint. The major steps that were used in fabrication of the CS3 chordwise splice specimen surface skin are as follows:

- Materials

  Hercules AS4/3502 G/E tape
  3M Co. AF147 350^{o}F adhesive - .03PSF unsupported
  American cyanamid FM73 250^{o}F adhesive - .06 PSF
  6AL-4V titanium

- Tooling

  Tool to outer mold line
  Use inner caul plate for hat runout mating surface
  Form titanium shims to 3^{o} break

- Cure

  Use cure cycle shown in Figure 71.

The major steps in the fabrication plan for hat-section stiffener segment for the CS3 chordwise splice specimen were as follows:

- Material

  Hercules AS4/3502 G/E tape

- Tooling

  Female mold - machined from aluminum
  Hard edge caul plates - to form flange faying surface
  Formed rubber center plug
  Templates for ply layup

- Processing

  Debulk each 20 plies under vacuum
  Cure using same procedure as used for MV5 hat section stiffeners

The major steps that were used in assembly of the CS3 chordwise splice specimen were as following:

- Drill metering holes in skin
- Prepare hat and skin surface for bonding
- Bond hat to skin
- Form rib clip
- Assemble details at joint
- Drill chordwise joint holes
- Complete assembly of specimen

One of the two hat runout specimens shown in Figure 87 has the bolt holes drilled through the titanium laminated joint. The photograph shows inner view of the joint specimen after the two hat runout specimens have been bolted together. A fiberglass duct plate with a transfer cavity and the outer titanium sheet are attached to complete the LFC slot across the chordwise splice.

11.7 FABRICATION OF CORNER CHORDWISE SPAR JOINT

The fourth concept selection specimen, CS4, was selected to develop a typical chordwise spar cap splice. The design for splicing the spar caps is similar to that for the cover splice in that centroid control is maintained and titanium is bonded into the graphite/epoxy at the joint, Figure 61. Since the spar caps are integrally manufactured with the covers, the titanium bonded into the spar caps must be compatible with the titanium bonded into the covers. The spar cap splice was designed by determining the load in the vertical-flange/effective-web and sizing the vertical-flange splice plates to carry the load from both members.

The fabrication plan for the CS4 concept selection specimen is as follows:

- o Tooling - use combination of CS2 and CS3
- o Tooling must be adaptable to CV4
- o All titanium was formed to net shape
- o Precured details were used

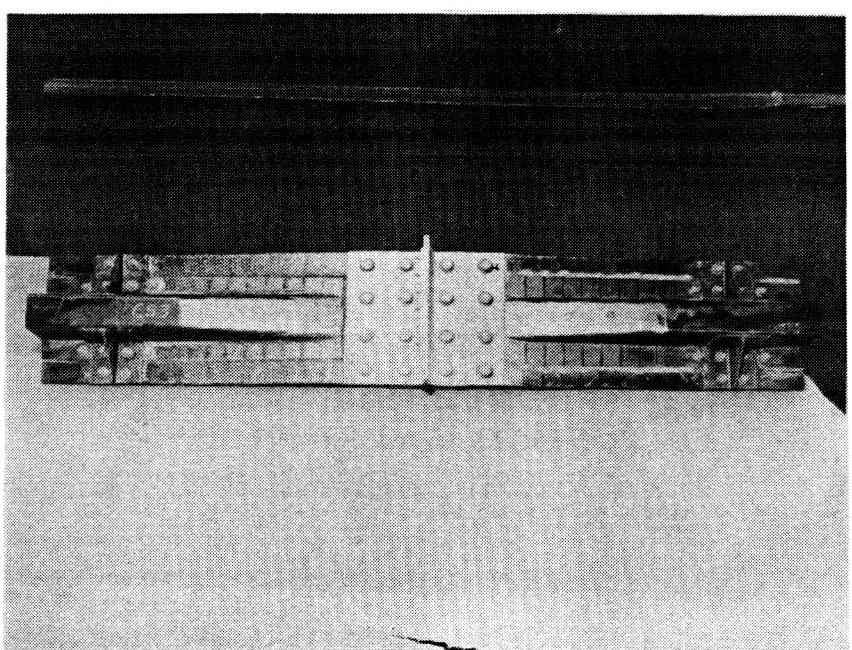


Figure 87. CS3 Chordwise Joint Assembly

12.0 TEST PROCEDURES AND EVALUATION OF RESULTS

Ancillary specimen and component tests were conducted to support and substantiate the design, analysis, and fabrication techniques for the wing surface structure of a 1993 LFC transport aircraft wing at approximately 55 percent semispan. Developmental tests were to be conducted in the following categories to establish the required assurance:

- Material verification
- Concept selection
- Concept verification

Of the above three categories, the first two categories were completed and are reported herein. The test procedures and evaluation of results are reported in the following subsections.

12.1 MATERIAL VERIFICATION SPECIMENS

Five groups of specimen tests were conducted to verify the performance of candidate materials in selected applications. Discussions of the test procedures and test results are presented in the following subsections for each of the five groups of specimens.

12.1.1 MV1 Lap Shear Specimens

Thirty single lap shear specimens, shown in Figure 47, were static tested to failure to evaluate two candidate adhesives for bonding titanium sheet to graphite-epoxy wing surface skin laminates. The primary objective of these tests was to establish the lowest acceptable adhesive cure temperature in order to minimize the residual stresses in the adhesive bondline resulting from the difference in the thermal coefficients of expansion of the two adherends as well as the flow characteristics of the adhesive during cure. The candidate adhesives evaluated were FM123-4 (American Cynamid) and EA9601.2 (Hysol Division of Dexter Corp.) whose densities were 0.045 lb/ft^2. The two curing temperature investigated were 180°F and 200°F. The test conditions were -65°F, room temperature, and +160°F with both wet and dry specimens.

In preparation of the lap shear specimens for testing, loading tabs were bonded to both ends of each test specimen to minimize the introduction of eccentric loads into the single overlap joint as shown in Figure 47. The results of the 30 lap shear specimens are presented in Table 16. The criteria for selecting the adhesive were: (1) flow characteristics, (2) bondline thickness variability, (3) bondline strength, and (4) failure mode desired to be predominantly graphite-epoxy laminate delamination. Upon completing the lap shear tests in addition to an evaluation of adhesive flow characteristics developed with bonded panel specimens, the FM123-4 adhesive cured at 200°F was selected over the EA9601.2 adhesive using the aforementioned criteria.

NUMBER OF SPECIMENS	ADHESIVE MATERIAL	CURE TEMP. (°F)	TEST TEMP. (°F)	SPECIMEN EXPOSURE	AVERAGE MAXIMUM STRESS (KSI)
5	FM123-4	180	R.T.	DRY	4.6
5	FM123-4	200	R.T.	DRY	5.1
5	EA9601.2	180	R.T.	DRY	4.9
5	EA9601.2	200	R.T.	DRY	4.9
5	FM123-4	200	(-65)	DRY	2.5
5	FM123-4	200	(160)	WET	1.5

TABLE 16.
LAP SHEAR SPECIMEN TESTS

12.1.2 MV2 Surface Element Tests

Surface element specimens with titanium sheet bonded to graphite-epoxy wing skins having LFC ducts and metering holes were fabricated and tested. The specimen drawing is shown in Figure 50. A slot duct was molded along the longitudinal centerline of the graphite-epoxy wing skin laminate during fabrication of the laminate followed by drilling metering holes through the base of the slot duct. Then the titanium sheet was adhesive bonded to the graphite-epoxy laminate with FM123-4 adhesive. After curing the adhesive, a slot was sawed in the titanium sheet along the slot duct centerline. A total of 23 specimens were fabricated and tested. Twenty-four specimens were planned to be tested, but one specimen was damaged during manufacture and was discarded.

Each specimen was mounted in a special test fixture which in turn was installed in a universal testing machine for compressive testing to failure. Prior to testing, selected specimens were instrumented with axial strain gauges located as follows:

(1) Mid-length of the specimen directly opposite the slot duct in the laminate

(2) Mid-length of the specimen on the graphite-epoxy laminate and half way between an edge of the specimen and the slot duct centerline

(3) Mid-length of the specimen on the titanium sheet and half way between an edge of the specimen and the centerline of the slot duct.

All specimens were tested in a 400-kip universal testing machine having a load range of 80 kips with an accuracy of $\pm$0.5 percent of indicated load or $\pm$0.1 percent of the load range whichever is greater. The strain indicator was a model SGE 062 whose accuracy is $\pm 5\ \mu$ in/in or $\pm$0.5 percent of indicated strain, whichever is greater. The specimens tested at temperature were monitored with a model TMP 206 recorder with an accuracy of $\pm 5^{\circ}$F.

The test results from the 23 specimen tests are summarized in Table 17. Twelve of the specimens were fabricated using the AS4/3502 graphite-epoyx material and the remaining 11 specimens were fabricated using T300/5208 graphite-epoxy material. In Table 17, the average initial failure loads and average maximum strains at failure are shown for each specimen group. Three specimens, fabricated with each of the two graphite-epoxy materials, were static tested to failure at room temperature and the remaining 17 specimens were environmentally conditioned and static tested to failure. The inital failure in all tests, with the exception of one, was disbond of the titanium sheet from the graphite-epoxy laminate. Test results were compared for the two graphite-epoxy materials and for each of the four test conditions. The comparisons are summarized in Table 18 and they are shown as percent increases and decreases in average initial failure loads and average maximum strains at failure. The percent changes are based on the test results from the specimens fabricated with the T300/5208 DV graphite-epoxy material. Referring to Table 18, the AS4/3502 graphite-epoxy material in the majority of tests performed better than the T300/5208 DV material. Thus, all subsequent test specimens and components were fabricated with the AS4/3502 graphite-epoxy material with the exception of the MV3 specimen and a portion of the MV4 specimens.

NUMBER OF SPECIMENS	GRAPHITE-EPOXY MATERIAL	SPECIMEN EXPOSURE	TEST TEMP. (°F)	AVERAGE INITIAL FAILURE LOAD (LB)	AVERAGE MAX STRAIN AT FAILURE (μIN./IN.)
3	T300/5208DV	DRY	R.T..	48,733	6,450
3	AS4/3502	DRY	R.T.	55,000	6,265
3	T300/5208DV	DRY	- 65	57,533	7,497
3	AS4/3502	DRY	- 65	73,767	8,456
2	T300/5208DV	COLD WET	- 65	60,600	8,130
3	AS4/3502	COLD WET	- 65	76,967	11,013
3	T300/5208DV	HOT WET	+160	56,467	10,470
3	AS4/3502	HOT WET	+160	62,600	10,547

TABLE 17.
SUMMARY OF AVERAGE MV2 SPECIMEN TEST RESULTS

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GRAPHITE EPOXY MATERIAL	SPECIMEN EXPOSURE	TEST TEMP. (°F)	AVG INITIAL FAILURE LOAD (LB)	PERCENT CHANGE IN FAIL. LOAD (%)	AVG MAX STRAIN AT FAILURE (μIN./IN.)	PERCENT CHANGE IN MAX STRAIN (%)
T300/5208DV	DRY	R.T.	48,733	12.9	6,450	-2.9
AS4/3502	DRY	R.T.	55,000		6,265	
T300/5208DV	DRY	-65	57,533	28.2	7,497	12.9
AS4/3502	DRY	-65	73,767		8,465	
T300/5208DV	COLD WET	-65	60,600	27.0	8,130	35.5
AS4/3502	COLD WET	-65	76,967		11,013	
T300/5208DV	HOT WET	+160	56,467	10.9	10.480	0.7
AS4/3502	HOT WET	+160	62,600		10,547	

NOTE:
(1) THE PERCENT CHANGES IN AVERAGE INITIAL FAILURE LOAD AND MAXIMUM STRAIN AT FAILURE ARE BASED ON THE T300/5208DV MATERIAL RESULTS.

TABLE 18.
COMPARISON OF MV2 SPECIMEN TEST RESULTS

12.1.3 MV3 Surface Element with Slot Duct Fatigue Test

An LFC surface skin element specimen having a slot duct region that is inclined at an angle to the specimen load axis, so that shear loads are introduced into slot-duct region containing the metering holes, was fabricated and fatigue tested. The MV3 test specimen was fabricated using T300/5208 DV graphite-epoxy tape material. Aluminum and fiberglass loading tabs were bonded and mechanically fastened to both ends of the specimen to facilitate testing. Figure 88 shows the instrumented test region of the test specimen. The fatigue loads test spectrum applied to the test specimen was representative of those loads that would occur in the upper wing surface of the 1993 LFC transport at the 55 percent wing semispan location. The spectrum consists of 219,404 load cycles in a lifetime which is representative of 90,000 flight hours.

The test specimen was instrumented with six pairs of axial strain gages and three pairs of strain rosettes, all installed back-to-back, as shown on Figure 88. In addition, two pairs of acoustic emission transducers were located on the specimen approximately 16.00 in from each end of the specimen. For testing, the instrumented MV3 specimen was installed in a lateral support assembly, and then the specimen/lateral support assembly was mounted in the cyclic testing machine.

The following sequence of test loads was applied to the MV3 test specimen:

(1) Two lifetimes of cyclic loads were applied at room temperature in accordance with the fatigue loads spectrum identified previously. One lifetime represents 219,404 load cycles.

(2) Two lifetimes of cyclic loads as described in (1) above, but increased by 20 percent, were applied at room temperature.

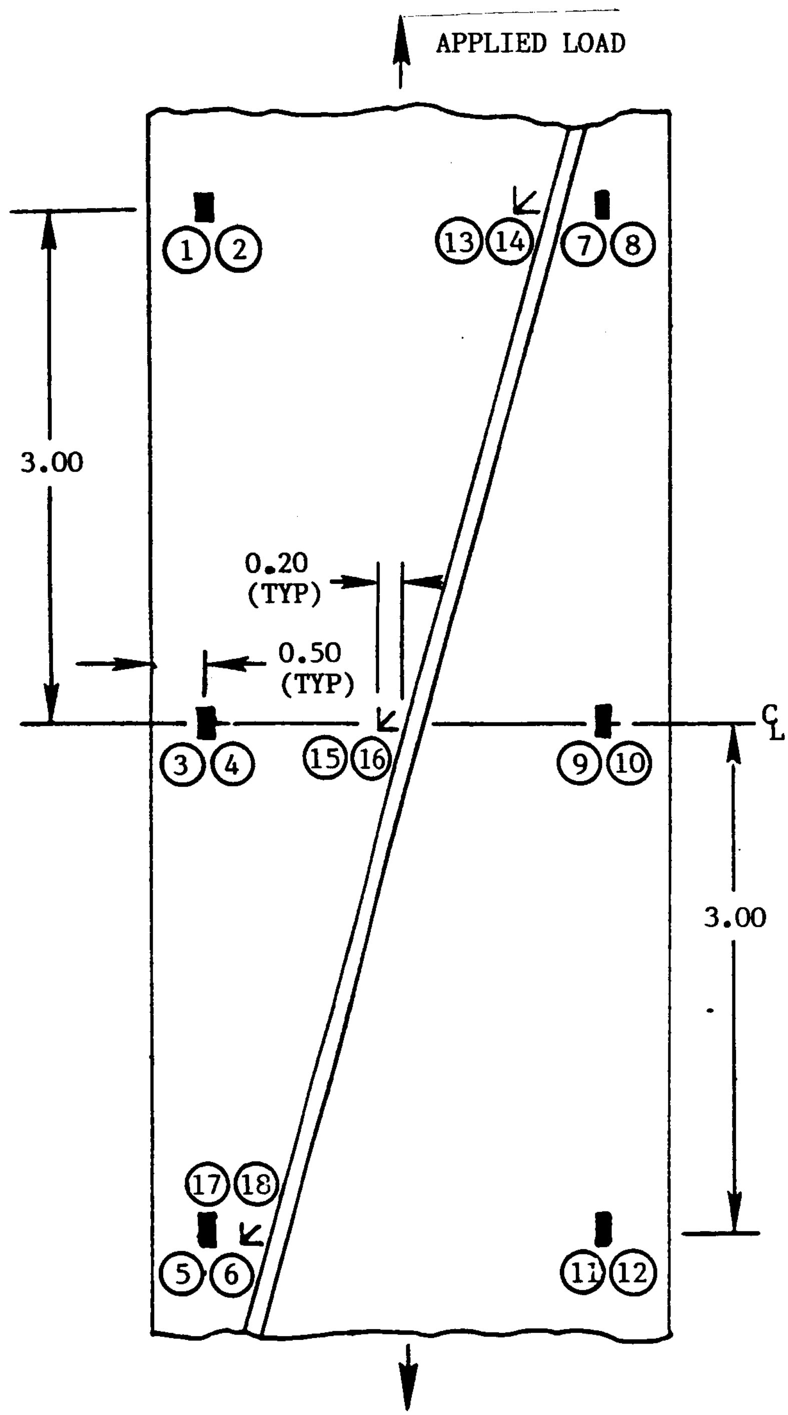

Figure 88. Instrumented Test Section of MV3 Specimen

(3) A static compression load was applied after the MV3 test specimen was conditioned dry to $-65^{\circ}F$. The static compression load applied to the specimen was the maximum load in fatigue loads spectrum identified in (1) above, but increased by a factor of 1.2 to a magnitude of -25,856 pounds.

(4) Twelve additional lifetimes of cyclic test loads were applied to the specimen using the loads spectrum defined in (2) above.

Prior to beginning the fatigue test, a displacement and strain survey was made. During the fatigue test, displacement and strain data were recorded upon the completion of each one-half lifetime up to and including four lifetimes. In addition to the displacement and strain surveys, the MV3 specimen test section was X-rayed after each one-half lifetime through three lifetimes and after the fourth lifetime. The radiographic film developed after the first lifetime showed several delaminations less than one-half inch in length in the vicinity of the slot duct and at approximately mid-length of the test section. Subsequent X-rays showed no apparent increase in the damage. By the end of the eighth lifetime, all axial strain gages were inoperative and one channel of one of the six strain rosettes was inoperative. The specimen did not fail upon completion of 16 lifetimes of cyclic loads and the test was suspended.

12.1.4 MV4 Single Hat-Stiffened Surface Specimen Tests

Single hat-section stiffened LFC wing surface element specimens were fabricated and tested to evaluate the processes for fabrication of the graphite-epoxy hat-section stiffener, and bonding of the hat-section stiffener to the representative graphite-epoxy wing surface skin laminate. A slot duct was molded along the lengthwise centerline of the wing surface skin laminate, and metering holes were drilled through the base of the slot duct. The specimen assembly was completed by bonding the titanium face sheet to the specimen subassembly and then machining the slot in the titanium face sheet over the slot duct. Figure 89 is a drawing of the MV4 single hat-stiffened wing surface element specimen.

Thirteen MV4 test specimens were fabricated and tested. The initial ten test specimens were fabricated with T300/5208 DV graphite-epoxy material and the remaining three specimens were fabricated with AS4/3502 graphite-epoxy material. Three candidate adhesives were used in bonding hat-section stiffener to the wing surface skin element. Two specimens were bonded with each of the three candidate adhesives. These initial six specimens were static tested to failure at room temperature in a compressive mode. The candidate adhesives used in assembly of the initital six specimens were FM73 (American Cynamid), EA9628 (Hysol Division of Dexter Corp), and AF163-2 (3M Company) film adhesives. Prior to testing, the ends of the MV4 specimens were potted in steel end frames and the specimen ends were machined flat and parallel. Also, the specimens were instrumented with axial strain gauges located at the mid-length and at the quarter-span length from one end of each specimen. In addition, three of the initial six specimens were acoustic emission (AE) monitored during the static tests. Two AE transducers were located on the titanium face sheet approximately 2.50 in from each end of the three MV4 specimens.

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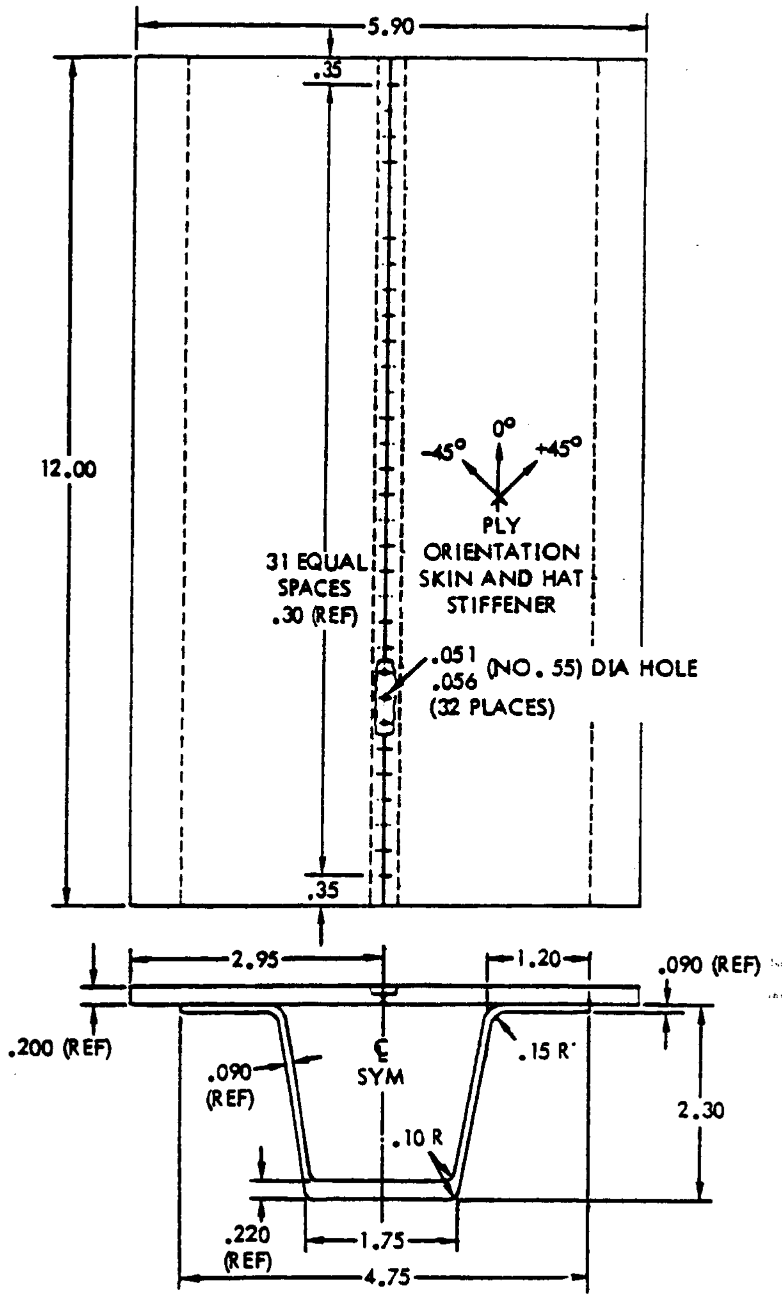

Figure 89. MV4 Test Specimen

After completing the initial six static tests, the results were reviewed and one of the three candidate adhesives was selected for fabrication of the remaining MV4 test specimens. The average test results are presented in Table 19. Referring to Table 19, neither of the three candidate adhesives showed a definite advantage over the other adhesives. The FM73 adhesive was selected for fabrication of additional MV4 specimens and future program specimens and components based on its usage in the previous phase of this program as well as being selected as the best adhesive from a survey on adhesives for metal-to-metal bonds in an Air Force program.

Four additional MV4 test specimens were fabricated and tested. Two of these specimens were static tested to failure in a compressive mode after having been conditioned dry to -65°F. The average failure load of these two specimens was 99,750 pounds. No initial delaminations of the titanium face sheet/graphite-epoxy laminate adhesive bonds occurred in these tests as occurred in the six MV4 specimens tested at room temperature previously discussed.

The third additional MV4 test specimen was conditioned in an environmental chamber at 160°F and 95-100 percent relative humidity until the moisture content was approximately one percent. Then the specimen was static tested to failure in a compressive mode at 160°F. Failure occurred at 80,000 pounds and the maximum strain achieved at failure was 8100 microin/in. The titanium face sheet separated from the graphite-epoxy laminate and the top end of the specimen "broomed".

The fourth additional MV4 specimen was fatigue tested at 160°F after being conditioned at 160°F and 95-100 percent relative humidity until each specimen contained approximately one percent moisture. The constant amplitude compressive load spectrum consisted of 200,000 load cycles, a maximum gross compression stress, and a stress ratio, R of +10.0. The fatigue life goal was four lifetimes (i.e., 800,000 load cycles). Fatigue testing of the MV4 specimen commenced and after accumulating 22,900 cycles, the test was suspended because a command versus measured load error limit was exceeded. Examination of the lower end of the MV4 specimen revealed extensive fretting damage and the slotted titanium face sheet was disbonded over approximately 25 percent of the total

NUMBER OF SPECIMENS	ADHESIVE	TEST TEMP. (°F)	AVERAGE INITIAL FAILURE LOAD (LB)	AVERAGE FINAL FAILURE LOAD (LB)
2	FM73	R.T.	78,500	114,250
2	EA9628	R.T.	81,000	103,750
2	AF163-2	R.T.	79,250	108,250

TABLE 19.
ROOM TEMPERATURE MV4 SPECIMEN TEST RESULTS

bonded area. The specimen was repaired and cyclic testing was continued after stabilizing the temperature at 160°F. Upon completing four lifetimes (800,000 cycles) of fatigue testing, the specimen was residual strength tested to failure at 160°F. The specimen failed in compression at 116,000 pounds and at a maximum strain of approximately 8700 microin/in. At failure, the slotted titanium face sheet separated on both sides of the slot. In addition, the graphite-epoxy skin and hat-section stiffener failed approximately two inches from the specimen centerline.

The remaining three MV4 specimens were removed from a MV5 multi-hat-stiffened specimen for the purpose of evaluating the titanium face sheet/graphite-epoxy laminate adhesive bond subjected to a hot-wet environment and then cyclic loaded. The specimens removed from the MV5 multi-test-stiffened-specimen were 20 in long, whereas the ten original specimens were 12 in long.

Two of the three MV4 specimens removed from the MV5 multi-hat-stiffened specimen were fatigue tested to the same fatigue loads spectrum and environment as the previously described MV4 fatigue test specimen. The two specimen tests were suspended after accumulating 400,000 and 50,000 load cycles. In both tests, the slotted titanium face sheet disbonded from the graphite-epoxy laminate. No additional testing was conducted on these two MV4 specimens.

It was realized that the constant amplitude fatigue loads spectrum applied in the above described three MV4 hot-wet tests was severe since the maximum load in the constant amplitude spectrum was the design limit load. Thus, the fatigue loads spectrum was revised as follows:

(1) Apply 55 percent limit compressive load to the MV4 specimen at room temperature and conduct a strain survey. Limit compressive load is 63.6 kips.

(2) Increase the specimen temperature to 160°F coincident with application of 4,999 cycles of compression-compression loads, R = +10.0, and a maximum load equal to 55 percent limit load.

(3) Increase the maximum load in each load cycle to 75 percent of limit compressive load, R = +10.0, and apply 500 load cycles at 160°F.

(4) Reduce the maximum load in each load cycle to 55 percent of limit compressive load, R = +10.0, and apply 5000 load cycles during which the specimen temperature is reduced from 160°F to room temperature.

(5) Continue cycling with the maximum load in each cycle equal to 55 percent limit load, R = +10.0, and room temperature for 189,499 cycles.

(6) Complete the lifetime of cyclic loads with application of one cycle of limit compressive load at room temperature during which a strain survey will be accomplished.

(7) Repeat above steps for three times to achieve four lifetimes of testing the MV4 specimen.

The third ~~MV4 specimen~~ removed from the MV5 multi-hat-stiffened specimen was fatigue tested to the above revised spectrum. After conditioning this specimen similarly to the previous three hot-wet fatigue specimens, the specimen was stabilized at 160°F and fatigue tested. A total of 800,000 cycles (four lifetimes) were accumulated without incident. Thus, the objective of accumulating four lifetimes of fatigue testing without a disbond failure of the titanium face sheet/graphite-epoxy bondline was successfully achieved.

12.1.5 MV5 Multi-Hat-Stiffened Surface Specimen Tests

Multi-hat-section-stiffened LFC wing surface panel specimens were fabricated and tested to verify the manufacturing processes and materials for use in fabrication of an LFC wing surface panel. The MV5 multi-hat-stiffened panel specimens had a configuration similar to the MV4 single hat-stiffened specimen except the MV5 specimens had three hat-section stiffeners. Figure 53 is a sketch of the MV5 specimens. Both basic materials and adhesive bondlines in the MV5 multi-hat-section-stiffened specimens were evaluated for simulated environmental conditions, impact damage, and compression loading to failure.

A total of five MV5 test specimens was fabricated. Four of the five specimens were tested in the MV5 series of tests and the fifth specimen was sectioned into three MV4-type specimens which were tested as hot-wet fatigue specimens, and those tests are described in Section 12.1.4.

The following materials were used in fabrication of the MV5 specimens:

(1) AS4/3502 preimpregnated graphite-epoxy tape material was used in fabrication of the hat-section stiffeners and surface skin laminates.

(2) 0.016 in 6Al-4V annealed titanium sheet material was used in fabrication of the specimen face sheet.

(3) FM73 (American Cynamid) film adhesive was used to bond the graphite-epoxy hat-section stiffeners to the graphite-epoxy surface skin laminates.

(4) FM123-4 (American Cynamic) film adhesive was used to bond the titanium face sheets to the grapite-epoxy surface skin laminates.

Prior to testing the MV5 specimens, the ends of each specimen were potted in steel end frames and then they were machined flat and parallel. Each of the four MV5 specimens were instrumented with 18 axial strain gauges installed back-to-back. The gauges were located at the specimen mid-length and approximately 3.00 in from both ends of the specimen.

All four MV5 specimens were static tested to failure in a compressive mode. The test environments and specimen codes are defined in Table 20. The two MV5 specimens that were impacted had the simulated damage imposed using a 0.500 in diameter aluminum projectile fired from an air gun. Both MV5 specimens were impact damaged with the projectile fired at approximately 200 ft/sec. The MV5 specimen test results are summarized in Table 21. Referring to this table, the failure load of the non-impacted specimen tested at -65°F was approximately 3 percent greater than the non-impacted specimen tested at room temperature. Similarly, the failure load of the impact damaged specimen tested at -65°F was approximately 18 percent greater than the failure load of the impact damaged specimen tested at room temperature. In all four specimen tests, the initial failure was separation of the titanium face sheet from the graphite-epoxy skin laminate. This failure was followed by separation of the hat-section stiffeners from the graphite-epoxy skin laminate.

A suction test was conducted on the fourth MV5 specimen. In preparation for the suction test, the center hat-section stiffener duct and the slot duct were sealed at both ends of the specimen by the potting compound applied for use

SPECIMEN CODE	SPECIMEN ENVIRONMENT AND PROCEDURES
MV5-DNR	TESTED DRY AT ROOM TEMPERATURE
MV5-DIR	IMPACTED SPECIMEN WITH SPECIMEN SUBJECTED TO A COMPRESSION LOAD OF 191,000 POUNDS, AND TESTED DRY TO FAILURE AT ROOM TEMPERATURE
MV5-DNC	TESTED DRY AT -65°F
MV5-DIC	A SUCTION FLOW TEST WAS CONDUCTED AT ROOM TEMPERATURE FOLLOWED BY IMPACTING SPECIMEN WITH IT SUBJECTED TO 150,000 POUNDS COMPRESSIVE LOAD. THEN THE SPECIMEN WAS TESTED DRY TO FAILURE AT -65°F.

TABLE 20.
MV5 SPECIMEN CODES AND TEST ENVIRONMENTS

SPECIMEN CODE	SPECIMEN IMPACT	TEST TEMPERATURE (°F)	FAILURE LOAD (LB)
MV5-DNR	NO	ROOM TEMP	312,500
MV5-DIR	YES	ROOM TEMP	225,000
MV5-DNC	NO	-65	322,000
MV5-DIC	YES	-65	265,000

TABLE 21.
MV5 SPECIMEN TEST RESULTS

in conducting the static compression test. A hole was drilled through the potting compound that sealed one end of the center hat-section stiffener. A pneumatic fitting was attached to the potting compound surface over the hole and a suction line was attached to the fitting. A flowmeter was installed in the suction line for making flow measurements. A pressure tap was installed in the potting compound on each end of the center hat-section stiffener for measuring pressure losses during the suction flow tests.

A suction test was conducted and the results are presented in Figures 90 and 91. These results are particularly interesting since they provide data which relate directly to an area of concern which arose during the SC-2 development test conducted in the Leading Edge Flight Test Contract (NAS1-16219) (Reference 2). In particular, Figure 90 shows excellent agreement between predicted and measured slot plus metering orifice pressure losses, which was not achieved in the SC-2 development test. Thus, the results from the MV5 suction tests reinforce confidence in the current prediction method for ducting system pressure losses.

Figure 91 presents local slot flow variations obtained during the test. Results are shown for two flow levels at 1-in spacing intervals along the entire slot length, and over and between metering orifices at either end of the slot. At the higher flow rate shown in Figure 91, 10.6 SCFM (Standard Cubic Feet Per Minute), the local flow variation is within ± 5 percent across the specimen span. This scatter increases to about ± 20 percent superimposed upon an increase in local slot flow of about 60 percent from the inboard to the outboard end of the slot when the total flow rate is decreased to 4.6 SCFM. A comparable slot flow for the LFC 1993 transport at the cruise design point would be between the two flows shown, and the flow variation would be satisfactory. The local flow measurements shown in the lower portion of Figure 91 were taken, alternately, over and between metering orifice locations. These data show no correlation of local slot flow with metering orifice location. The results indicate satisfactory performance of the slot and metering orifice geometry as tested.

12.1.6 Summary of Material Verification Tests

The summary of the five groups of material verification tests is shown in Figure 92.

The MV1 single lap shear tests resulted in the selection of the FM123-4 adhesive cured at 200°F. The FM123-4 adhesive yielded higher strength with low flow characteristics.

The MV2 surface element compression tests resulted in the selection of the AS4/3502 graphite epoxy material. The AS4/3502 G/E material yielded higher strength with void free thick laminate.

The MV3 surface element fatigue test verified the fatigue life of the graphite skin with small metering holes. The MV3 specimen did not fail upon completion of 16 lifetimes of cyclic loads.

The MV4 single hat-stiffened surface specimen tests verified the ultimate strength and fatigue life of the FM73 adhesive bonds of the hats to skins.

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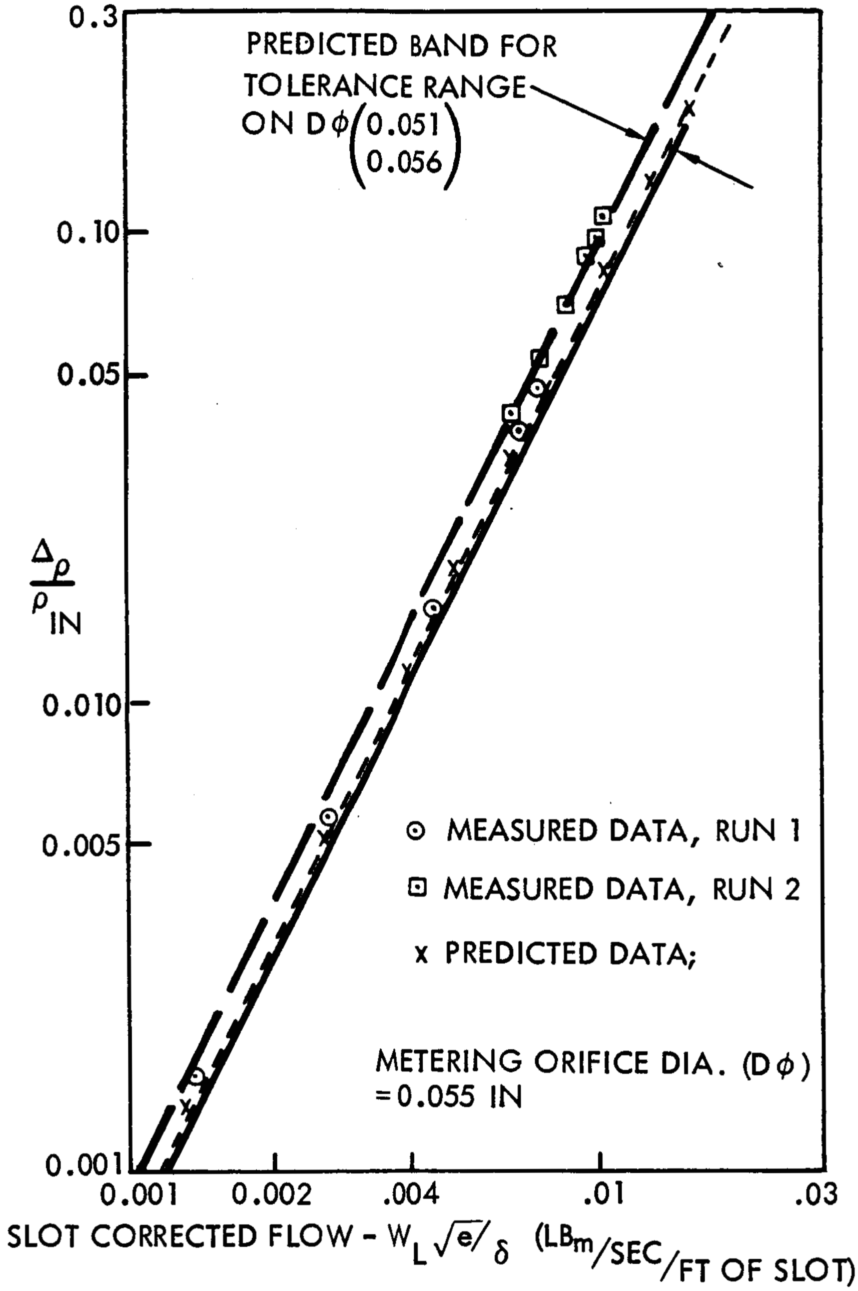

Figure 90. Comparison of Predicted and Measured Slot Plus Metering Orifice Pressure Losses for MV5 Specimen

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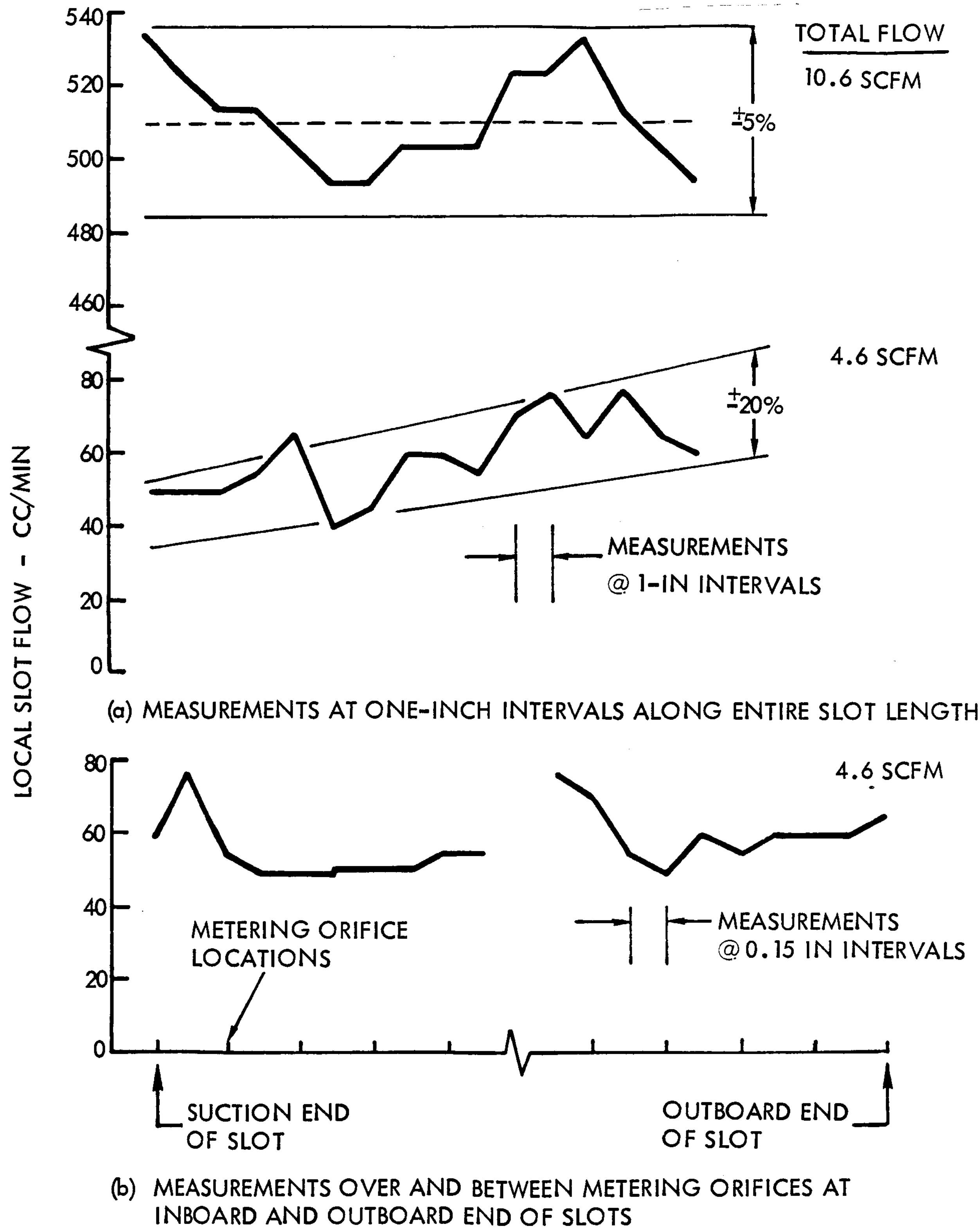

Figure 91. Local Slot Flow Variation Over MV5 Specimen

TEST	SELECTED MATERIALS	VERIFICATION
MV1	FM 123-4 CURED AT 200°F	o LOW FLOW o HIGH STRENGTH
MV2	AS4/3502 CURE CYCLE	o VOID FREE o HIGH STRENGTH
MV3	GRAPHITE SKIN METERING HOLE	o FATIGUE o ULTIMATE
MV4	GRAPHITE HAT/SKIN FM73	o FATIGUE o ULTIMATE
MV5	PANEL	o ULTIMATE o IMPACT o SUCTION

Figure 92. Summary of Material Verification Tests

The MV5 multi-hat-stiffened surface tests verified the complete surface panel for compression loads. The MV5 test also evaluated the panel resistance to impact damages. Suction flow measurements made on one specimen correlated very closely with predicted values.

12.2 CONCEPT SELECTION SPECIMEN

Four groups of concept selection (CS) specimens were used to develop the critical design details, tools, and manufacturing processes. Discussion of the test procedures and test results are presented in the following subsections for each of the four specimens:

- CS1 Rib duct to surface specimen
- CS2 Spar cap specimen
- CS3 Chordwise splice specimen
- CS4 Spar-Cap/chordwise splice specimen

12.2.1 CS1 Rib Cap Duct/Hat-Stiffened LFC Wing Surface Assembly Specimen

The primary objective of these tests was to evaluate the design and producibility of a chordwise rib-cap duct/hat-stiffened LFC wing surface assembly. The test specimens were designed to satisfy the LFC and structural requirements of the 1993 LFC transport wing upper surface at its 55 percent semispan. The design condition for the CS1 test specimen was a static tension load applied to the rib-cap web combined with fuel pressure load against the rib-cap duct.

Figure 54 shows a sketch of the CS1 specimen. Figure 55 shows the first CS1 specimen.

The design ultimate load for the tension pull-off condition was 21,600 lb, and the simulated fuel pressure applied to the rib-cap duct walls ranged from 0 to 8.0 psi.

In preparation for static testing, the CS1-S-1 specimen skin surface was adhesive bonded to a 1.0 in thick aluminum plate which, in turn, was clamped to the lower platen of the universal testing machine. The 1.0 in aluminum plate had three machined slots for reduction of the plate stiffness across its width to provide for an accurate load introduction into the specimen. In addition, the periphery of the specimen was clamped to the aluminum plate to minimize peel stresses in the specimen/aluminum plate bondline. The pull-off test load was introduced into the specimen through an aluminum strap mechanically attached to the rib-cap web. Tension load applied to the rib cap tends to separate it from the wing surface structure.

Test loads were applied to the specimen in increments of 4000 lb. Noises emanated from the specimen when the test load reached approximately 12,000 lb. These noises may have been the onset of disbonding and/or delaminations in the specimen as failure occurred at approximately 13,000 lb. The specimen failure mode was a shear failure in the adhesive bonds joining the structural elements to the rib-cap web. The specimen failure load was much lower than the design ultimate load of 21,600 lb. An investigation of the failed CS1-S-1 specimen revealed that the rib-cap web bonded surface was contaminated on the outside wall of the rib duct. As a result, a quality bond was not achieved during fabrication and the premature failure occurred. A second specimen, CS1-S-2, was fabricated and tested. The CS1-S-2 specimen failure load was 24,500 lb compared to a design ultimate load 21,600 lb. The failed specimen is shown in Figure 93.

12.2.2 CS2 Spar Cap and Single Lap Shear Joint Specimen

The primary objectives of these tests were to evaluate the spar cap design concept and the spar-cap to spar-web joint. Specifically, the spar-cap legs to which the spar web and the wing leading edge are attached were tested for compressive buckling stability. The single lap spar-cap to spar-web joint was static tested in a tensile-shear mode.

The spar cap specimen, CS2-1, is representative of the upper front spar cap of the 1993 LFC transport wing. It was designed to satisfy the design requirements of the transport wing at its 55 percent semispan. Figure 56 shows the CS2-1 spar cap specimen. As shown in this figure, the spar cap flange is integral with the upper wing surface cover and the flange on the test specimen ended at the flange/cover transition location.

The CS2-1 spar cap specimen was designed for a compression load in the specimen and the predicted failure load was approximately 163.6 kips.

Prior to compressive testing, both ends of the CS2-1 specimen were potted in steel end-frames using Magnabond 69-9, Parts A and B, potting compound.

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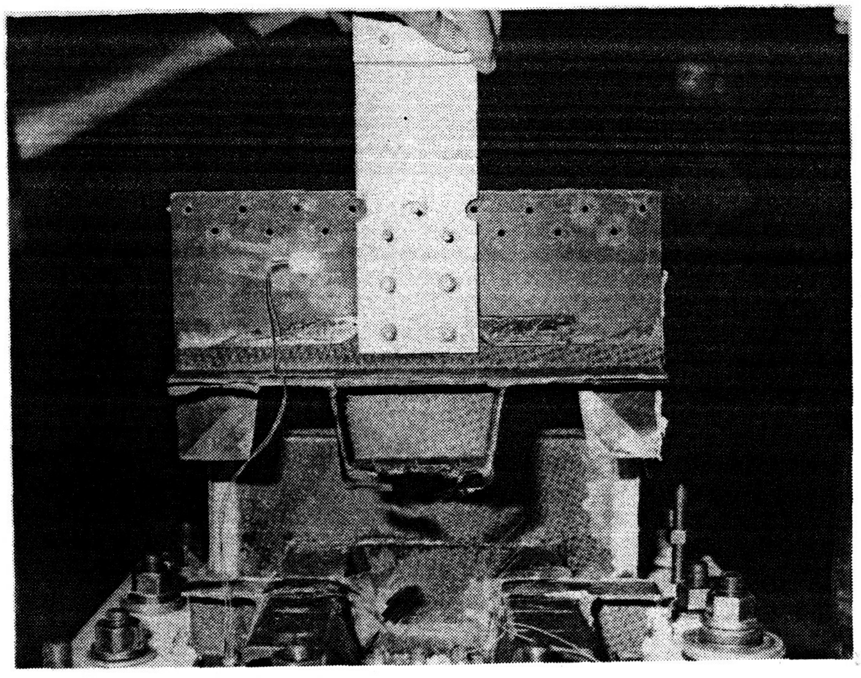


Figure 93. Failed CS1-S-2 Test Specimen

Compression loads were applied to the CS2-1 spar cap specimen in increments of 20 kips up to and including 100 kips. Thereafter, test loads were applied in increments of 10 kips until failure occurred. Failure occurred at 180 kips which was approximately 10 percent above the predicted failure load.

The single lap joint specimen is representative of the upper front spar-cap to spar-web joint of the 1993 LFC transport wing at its 55 percent semi-span location. Figure 57 shows the CS2-2 and -3 single lap shear joint specimens.

The CS2-2 and -3 single lap joint specimens were assembled with two 0.375 in mechanical fasteners. The predicted fastener bearing failure load on the spar web element was approximately 16.0 kips.

The CS2-2 and CS2-3 single lap joint specimens were static tested in a tensile-shear mode. Both specimens were tested in the 75-kip MTS testing machine. The CS2-2 specimen failed at 14,700 lb. A bearing failure initiated in the thinner part which was representative of the spar web. Final failure occurred as a net tension failure through one of the fastener holes of the thinner part of the joint specimen. The CS2-3 joint specimen failed at 14,040 lb. The failure mode was the same as the CS2-2 specimen with an initial bearing failure followed by a net tension failure through one hole. The specimen loads were approximately 90 percent of the predicted failure load.

12.2.3 CS3 Wing Surface Chordwise Joint Specimen

The objective of this test was to evaluate an LFC upper wing surface chordwise joint design concept. This chordwise joint was deemed necessary in the manufacture of the transport aircraft and was located coincident with a wing rib.

The wing chordwise joint specimen, CS3, is representative of an upper wing surface chordwise joint for the 1993 LFC transport. It was designed to satisfy the design requirements of the transport wing at its 55 percent semi-span. Figure 58 shows one-half of the chordwise joint specimen and it is symmetrical about the joint centerline. As shown in this figure, the hat-section stiffener tapers in height as it approaches the joint area. The joint is made using inner and outer titanium splice plates with high strength bolts fastening the splice plates to the CS3 specimen ends. Figure 59 shows a cross section through the CS3 specimen along the LFC slot centerline. Also, titanium interleaves are incorporated in the joint area of the specimen to provide sufficient bearing strength.

Figure 94 shows the hat-stiffener side of the CS3 wing chordwise joint specimen. Prior to compressive testing in a universal testing machine, both ends of the CS3 specimen were potted in steel end-frames using Magnabond 69-6, Parts A and B, potting compound.

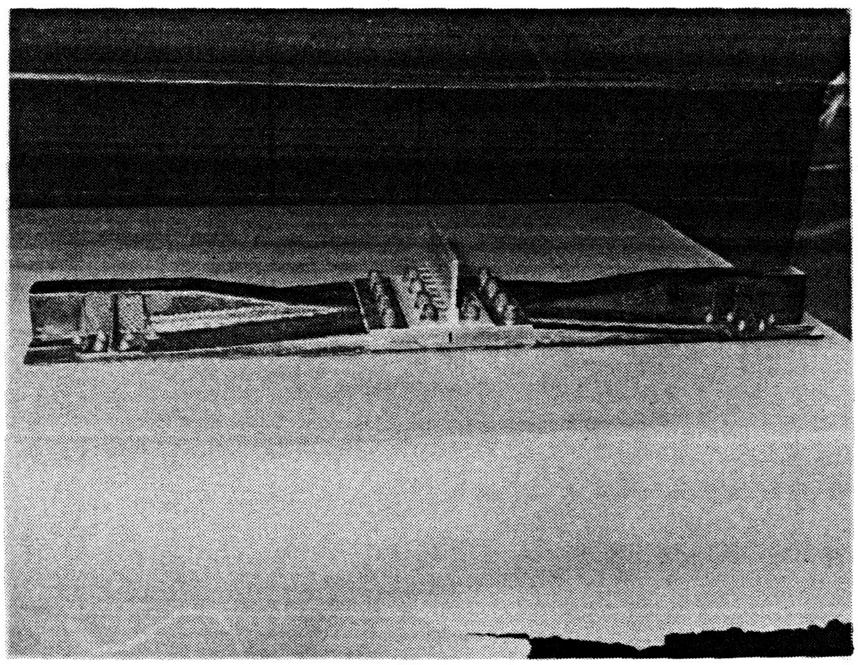


Figure 94. CS3 Specimen Ready for Preparation for Testing

Upon installation in the universal testing machine, lateral support was provided at the centerline of the joint to represent the wing rib support in the aircraft wing. During manufacture of the specimen, aluminum angles were attached to the specimen's inside splice plate using the rows of mechanical fasteners located closest to the specimen centerline for attachment of the lateral support structure.

Compression loads were applied to the CS3 specimen in increments of 10 kips in the 400-kip universal testing machine. Deflection data were recorded at each increment up to 80 kips. The dial gauges were removed and the loading increased until failure occurred at 119,000 lb. The failure load was 8 percent above the predicted failure load. Failure occurred near one end of the potted steel frames.

12.2.4 CS4 Surface-Chordwise Joint/Spar-Cap Splice Specimen

The principal objective of this test was to evaluate the wing surface chordwise joint design concept at the front spar intersection. This joint includes the front spar cap splice.

The wing chordwise specimen, CS4, was representative of an upper wing surface chordwise joint for the 1993 LFC transport. It was designed to satisfy the design requirements of the transport wing at its 55 percent semi-span. Figure 61 shows the spar-cap chordwise joint specimen.

Prior to compressive testing, the ends of the CS4 specimen were potted in steel end-frames using Magnabond 69-6, parts A and B, potting compound. Lateral support provisions were included at the mid-length of the specimen to preclude a general instability failure during the compression test. Figure 95 shows the instrumented CS4 specimen installed in the universal testing machine with the lateral support in place.

Compression loads were applied to the CS4 specimen in increments of 20 kips up to 140 kips and 10 kips above 140 kips in the 400-kip universal testing machine. Deflection data were recorded at each increment up to 150 kips. The dial gauges were removed and the loading increased until failure occurred at 154,000 lb. The failure load was 12 percent below the predicted failure load. Failure occurred near one end of the potted steel frames.

12.2.5 Summary of Concept Selection Tests

The summary of the four groups of concept selection tests is shown in Figure 96. The CS1 chordwise rib-cap duct/hat-stiffened LFC wing surface assembly was ultimate tested for the critical tension pull-off condition. The failure load for CS1 specimen test exceeded the predicted failure load by approximately 13.3 percent.

The CS2-1 spar cap was tested for compressive buckling stability. The CS2-1 specimen failed at approximately 10 percent higher load than predicted.

Two CS2-2, -3 single lap specimens representing the spar-cap leg to spar web joint were tested to failure in a tensile-shear mode. The fastener bearing failure stress was approximately 90 percent of that used in the wing analysis.

The CS3 wing surface chordwise joint specimen was ultimate compressive tested. The failure load was 8 percent above the predicted failure load. The CS4 surface-chordwise joint/spar cap splice was ultimate compressive tested. The failure load was 12 percent below the predicted failure load.

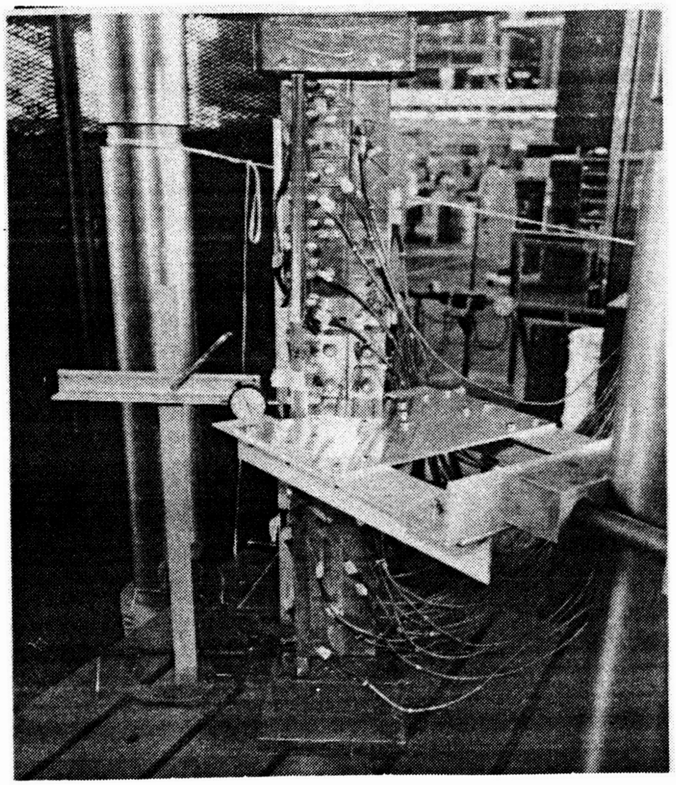


Figure 95. Instrumented CS4 Specimen Installed in Universal Testing Machine

TEST	SPECIMEN	LOAD PREDICTED/FAILURE (PERCENT)
CS1	RIB DUCT/SURFACE	+13.3
CS2-1	SPAR CAP	+10.
CS2-2,-3	SPAR CAP/SPAR WEB JOINT	-10.
CS3	CHORDWISE SPLICE	+ 8.
CS4	SPAR-CAP/CHORDWISE SPLICE	-12.

Figure 96. Summary of Concept Selection Tests

13.0 COST OF LFC COMPOSITE WING STRUCTURE

During Phase I of Reference 1 Lockheed conducted a comprehensive system study to evaluate the advantages of Laminar Flow Control (LFC) for future transport aircraft in the 1985-1995 time period. The study showed the use of LFC resulted in significant reductions of aircraft weight, fuel consumption and direct operating costs.

Investigations were conducted to determine the optimum configuration for a 400-passenger long-range transport featuring LFC and, as a baseline for comparison, a similar aircraft without LFC. The two aircraft configurations were optimized for the same mission, defined by a 84,800 lb payload, a range of 6500 n mi at cruise M = 0.80 and 10,000 ft field length. Both aircraft included advanced technology applications such as supercritical airfoil shapes, active controls, and composite primary and secondary structures.

The optimum configuration of the long-range 1993 transport aircraft without the LFC system is illustrated in Figure 97. The design features a wide-body fuselage, low wing, low horizontal stabilizer and four plyon-mounted engines beneath the wing. The wing has a span of 246.7 ft and contains all of the mission fuel in the wing box structure. The 226-ft long fuselage is sized to accommodate a typical 10/90 passenger mix with 40 in first class, seated 6 abreast, and 362 in tourist, seated 10 abreast. Space allowances are made for galleys, lavatories, closets, cabin crew provisions and rest areas for flight crews. Space for LD-3 cargo containers is provided in the underfloor area forward of the wing box and aft of the landing gear compartment. A bulk cargo bay is also provided at the rear of the pressurized belly. These cargo bays will accommodate 37,000 lb of cargo.

The optimum configuration of the long-range 1993 transport with LFC has engines mounted on pylons extending from the rear fuselage, Figure 98. This location provides a clean wing for the LFC suction system. The fuselage is similar to that of the non-LFC design except the length is increased to 240 ft to accommodate engine mounting structure aft of the passenger cabin. A "tee-tail" configuration is used with the rear-mounted engines.

LFC suction capability is provided for the upper and lower surfaces of wing and horizontal stabilizer. An independently driven suction pump for the LFC system is located under each wing root.

The LFC and turbulent aircraft baseline costs have been estimated by utilization of parametric/cost models based on historical data and similarity with other parts of known cost. Cost estimates for the LFC and turbulent aircraft configurations studied for this program were generated by the Lockheed-Georgia Company's Acquisition Cost Program. This is a parametric program based on actual costs of existing aircraft systems. Aircraft variables such as aircraft configuration, structural weight, gross weight, fuselage volume, speed, and range were considered when estimating the LFC and turbulent aircraft system costs. Economic variables were identical for both the LFC and turbulent configurations to ensure consistent comparisons. The costing ground rules used in this study are listed as follows:

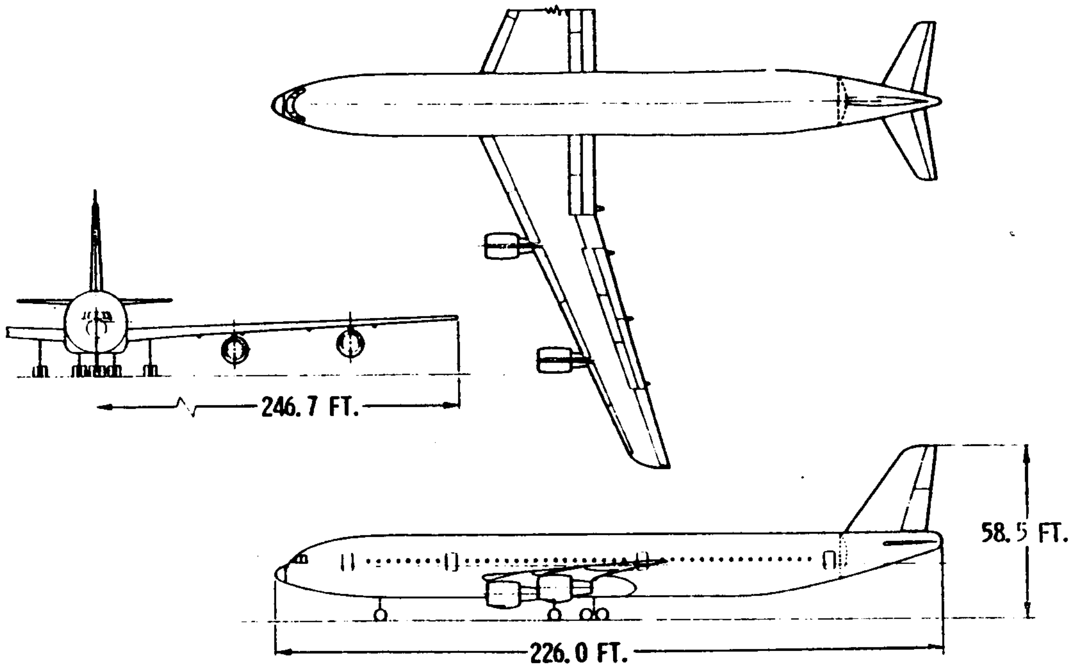


Figure 97. Equivalent 1993 Transport Without LFC

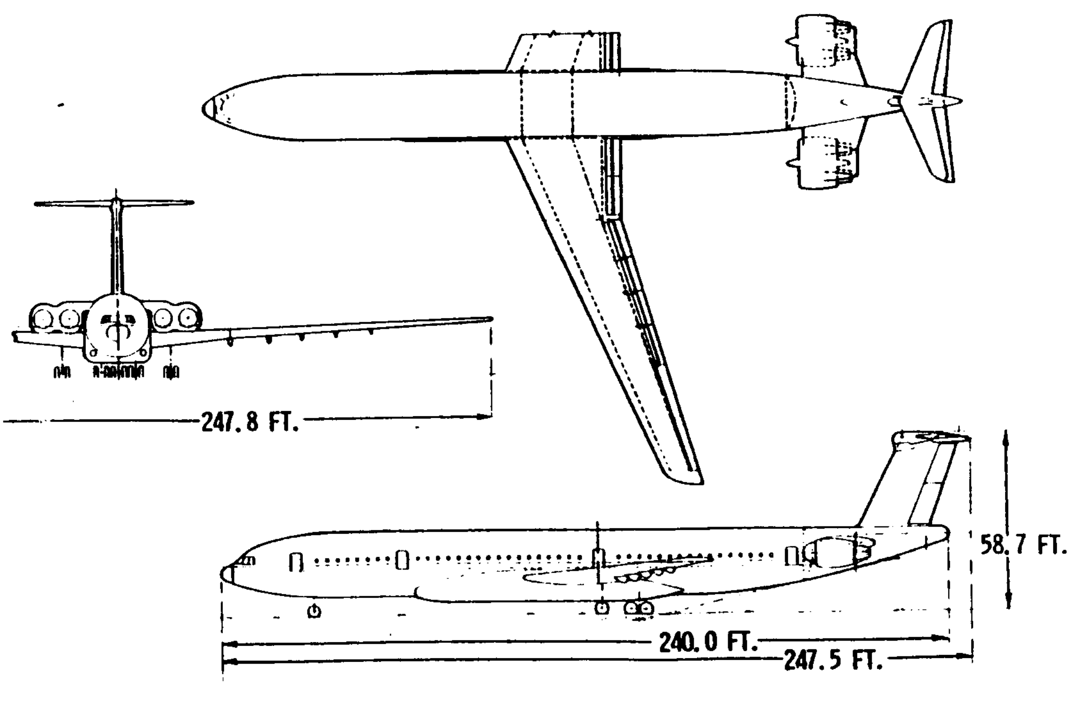


Figure 98. 1993 LFC Transport

- o 1981 Dollars
- o 350 Production Units
- o Maximum 4 Units/Months
- o Comparable Technology
- o 1993 Graphite/Epoxy
- o New Design for this Mission

Except for the wing-mounted engines of the turbulent design, the planforms of the LFC and non-LFC wings are very similar. The planforms and comparative data for the two wings are shown in Figure 99.

Leading-edge slats are provided on the turbulent wing. However, they are not required on the LFC wing. Trailing-edge flaps, spoilers, and ailerons are similar for both designs.

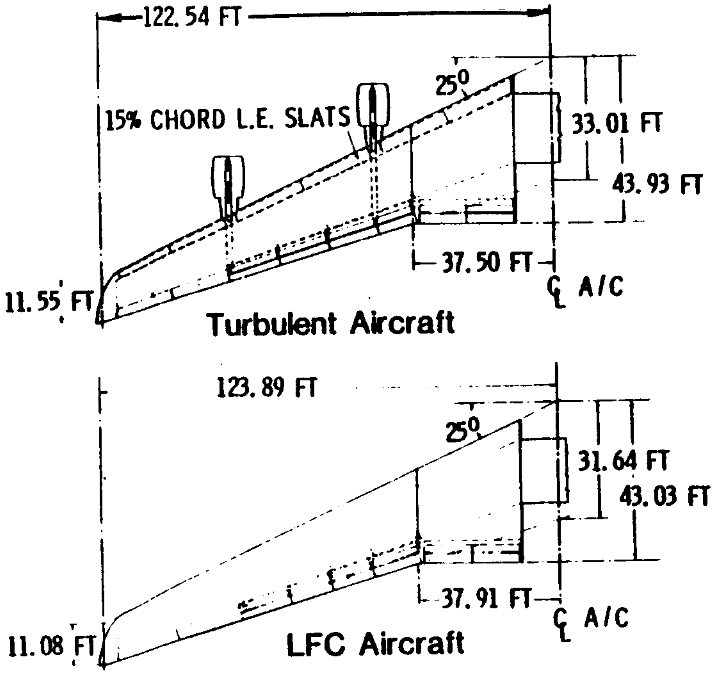
The estimated cumulative average cost for the two wings were as follows:

Turbulent wing	$12,116,000
LFC wing	11,074,000

Sustaining costs and profits should be added to the above production costs. Cost for fabricating the wing box is approximately 55 percent of the total wing. These costs are approximately equal to equivalent metal wings. The data show that the use of LFC permits a slight reduction in wing area, a slight increase in aspect ratio, and an increase in wing thickness, which results in a lower wing weight.

Comparison of the LFC and turbulent (non-LFC) aircraft Figure 100 shows that the application of a LFC system in the aircraft design permits a reduction in gross weight of 8.5 percent and a reduction in the mission fuel weight of 21.7 percent. Other items of interest tabulated in the weight comparison, Figure 100, illustrate the relatively small penalties imposed by LFC, such as the surface and system penalty of 0.6 percent of empty weight. This results from the efficiency of the integral-with-structure suction system which imposes a penalty of just 0.71 LB/FT^2. From these data, it can be seen that the effort expanded during this contract effort resulted in the design, development, and testing of a highly efficient LFC wing box structure.

Input of variables to the basic acquisition program would yield cost estimates for aircraft systems equivalent in technology to the data base in the program. Unmodified input data would result in acquisition cost estimates for a current state-of-the-art aircraft. This would represent a conventional aluminum aircraft with existing systems. Input data was therefore modified to reflect the cost effects of the new technology incorporated in both the LFC and turbulent aircraft configurations.



COMPARATIVE DATA - TURBULENT VS LFC

		TURBULENT A/C	LFC A/C
WING AREA	(BASIC) (BATTED)	5461.01 FT² 5870.53 FT²	5292.57 FT² 5724.12 FT²
WING A.R.	(BASIC) (BATTED)	11.0 10.232	11.6 10.725
WING TAPER RATIO		.35	.35
WING SPAN (THEORETICAL)		245.09 FT	247.78 FT
WING T/C RATIO		.0988	.1128
AIRCRAFT T.O.G.W.		645,073 LB	592,205 LB
MISSION FUEL (6500 N.MI)		274,164 LB	214,711 LB
WING WEIGHT		68,365 LB	62,100 LB
WING LFC WEIGHT PENALTY		0	5,152 LB

Figure 99. Wing Comparison 1993 Transport LFC vs Turbulent

	TURBULENT	LFC	% CHANGE
• BASIC WING AREA, FT^2	5,461	5,293	- 3.2
• ENGINE THRUST (EA.), LB	36,790	33,540	- 8.8
• WEIGHT EMPTY, LB.	252,478	253,885	+ 0.6
• GROSS WEIGHT, LB.	645,073	590,496	- 8.5
• FUEL, LB.	274,164	214,711	-21.7
• LFC PENALTY, LB. (NOT RESIZED)	-	9,607	+ 3.8

Figure 100. Characteristics Comparison (Approximate)

A baseline turbulent aircraft was estimated for comparison with the LFC aircraft being designed. The first step was to isolate the system and structure that will change due to the LFC design configuration. Cost increments peculiar to these LFC design concepts, i.e. surface slots, leading edge cleaning system, LFC suction system and ducts, rear mounted engines, titanium outer skins, etc. were isolated for use in demonstrating the cost impact of various parameter and design changes which will occur throughout the program.

Cost factors for the LFC designs were developed in detail down to a level sufficient to permit comparison with a baseline turbulent aircraft. The LFC acquisition cost penalties are summarized in Figure 101.

An acquisition cost comparison is shown for the turbulent (non-LFC) and LFC aircraft in Figures 102 and 103.

The LFC aircraft acquisition cost is $1,900,000 more than an equivalent technology turbulent aircraft, Figure 104. Fuel requirements for the LFC aircraft are shown to be significantly lower than those for the turbulent aircraft. Fuel requirements in terms of "seat statute mile per gallon" are shown in Reference 1, Figure 174 for various stage lengths in terms of statute miles. The incremental fuel costs based on $1.50 per gallon are calculated for two 3800 statute mile flights per day. The calculation shows that the lower fuel costs for LFC offset the higher incremental costs of LFC in less than six months. Fuel costs are shown to be approximately $4,000,000 per year lower for the LFC airplane.

DOLLARS PER AIRCRAFT IN MILLIONS	
LFC SUCTION SYSTEM, ENGINES, DUCTS, CONTROLS	2.03
LEADING EDGE CLEANING SYSTEM	0.40
SURFACE PANEL SLOTS, METERING HOLES, TITANIUM FACE SHEETS, MISCELLANEOUS	0.76
TOTAL	3.19

Figure 101. LFC Acquisition Cost Penalties

DOLLARS PER AIRCRAFT IN MILLIONS

• ACQUISITION COST PENALTY, LFC SYSTEM	+ 3.19
• ACQUISITION COST BENEFIT	- 1.29
- SMALLER WING	
- REDUCED ENGINE SIZE	
- NO LEADING EDGE DEVICES	
- LIGHTNING STRIKE	
- MISCELLANEOUS	
• TOTAL LFC PENALTY	+ 1.90

Figure 102. LFC Acquisition Cost Increment

	(1981 DOLLARS - MILLIONS)		
	TURBULENT	LFC	% CHANGE
RDT&E	3,040	3,170	+4.3
RECURRING, 350 AIRCRAFT	24,690	25,200	+2.1
TOTAL	27,730	28,370	
AVERAGE PRICE FOR 350	79.2	81.1	+2.4
COST INCREASE PER AIRCRAFT		1.9	

Figure 103. Acquisition Cost Comparison

	TURBULENT	LFC
COST INCREASE PER A/C	-	$1,900,000
SEAT MPG (REF. 1)	93	119
FUEL PER DAY (TWO 3800-MILE FLIGHTS)	32,700 GAL.	25,300 GAL.
COST PER DAY @ $1.50/GAL.	$49,000	$38,000
COST PER YEAR (360 DAYS)	$17,640,000	$13,680,000
INCREMENTAL FUEL COST PER YEAR	$ 3,960,000	-

Figure 104. Incremental Fuel Costs Projected for Turbulent versus LFC

14.0 CONCLUDING REMARKS

An LFC long range transport was defined during Phase I (Ref. 1) of the LFC Program. This transport is illustrated in Figure 1. An integrated LFC wing structural concept was identified and some components were tested for the wing for this transport. This wing was used as the baseline wing for continuing the development of the LFC wing surface. Structural loads and stiffnesses were calculated for use in preliminary design of the baseline wing.

An in-depth preliminary design of the baseline LFC wing was accomplished. Structural members were located and sized. The LFC suction surfaces and internal ducting were also located and sized. Detail design and verification of the surface panel were accomplished. The ancillary test plans, manufacturing processes and testing for all the material verification and concept selection specimens were completed. Materials were selected and verified by test. Detail design of the concept verification and concept demonstration panel was completed. Preliminary plans were made for testing concept verification and demonstration panels. The fabrication and testing of the concept verification and the demonstration panels was not accomplished under the WSSD project. These LFC development activities were deferred until some later initiative. Cost of the baseline LFC aircraft was estimated and compared to a non-LFC aircraft. Fuel costs are shown to be approximately $4,000,000 per year lower for the LFC aircraft. The calculation shows that the lower fuel costs for LFC offset the higher incremental costs of LFC in less than six months. The mission fuel weight was 21.7 percent lower for the LFC aircraft. The empty weight for the LFC aircraft was only 0.6 percent higher. This results from the efficiency of the integral-with-structure suction system which imposes a penalty of just 0.71 LB/FT^2. From these data, it can be seen that the development activities expanded during the contract effort continued the design, development, and tests of the highly efficient LFC wing box structure.

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15.0 ADVANCED DEVELOPMENT REQUIREMENTS

Major requirements for the development of LFC structure center around the use of advanced composite materials and characteristics peculiar to the fabrication of LFC surfaces. Included are the following:

(1) In the development of composite LFC wings, additional effort is required in investigating the main landing gear support area, chordwise joints, access panels, wing/fuselage joints, and hybrid LFC leading edge panels.

(2) Continued development of surface slotting procedures is required. Advances in laser and water jet techniques should be monitored to evaluate potential improvements leading to reduced slot widths and faster cuting rates.

In addition, a five-axis drive system should be developed for sawing slots. This system would track the slot and maintain exact saw-to-skin depth control reducing saw exposure to a minimum. Saw torque would be monitored and controlled by a microprocessor to allow stopping the saw prior to failure. After replacing a worn or broken saw, the computer system would return the saw to the exact spot where it stopped, with minimum disruption in the slot sawing process.

(3) In advanced materials development, powdered aluminum sheet materials should be considered as a candidate for the slotted outer surface. Powdered aluminum is corrosion resistant and no anodizing or corrosion protection would be required. Powdered titanium sheet material should be evaluated as a candidate for a porous outer skin.

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1. Report No.	2. Government Accession No.	3. Recipient's Catalog No.
NASA CR 172330		

4. Title and Subtitle	5. Report Date
DEVELOPMENT OF LAMINAR FLOW CONTROL WING SURFACE COMPOSITE STRUCTURES	May 1984
	6. Performing Organization Code
7. Author(s): L. B. Lineberger, et. al.	8. Performing Organization Report No.: LG84ER0035
	10. Work Unit No.
9. Performing Organization Name and Address: Lockheed-Georgia Company, 86 South Cobb Drive, Marietta, GA 30063	11. Contract or Grant No.: NAS1-17487
	13. Type of Report and Period Covered: Contractor Report
12. Sponsoring Agency Name and Address: National Aeronautics and Space Administration, Washington DC 20546	14. Sponsoring Agency Code

15. Supplementary Notes

Langley Contract Monitor: Dal V. Maddalon
Final Report

16. Abstract

The dramatic increases in fuel costs and the potential for periods of limited fuel availability provided the impetus to explore technologies to reduce transport aircraft fuel consumption. NASA sponsored the Aircraft Energy Efficiency (ACEE) program beginning in 1976 to develop technologies to improve fuel efficiency. This report documents the Lockheed-Georgia Company accomplishments under NAS1-16235 LFC Laminar-Flow-Control Wing Panel Structural Design And Development (WSSD); Design, manufacturing, and testing activities. An in-depth preliminary design of the baseline 1993 LFC wing was accomplished. A surface panel using the Lockheed graphite/epoxy integrated LFC wing box structural concept was designed. The concept was shown by analysis to be structurally efficient and cost effective. Critical details of the surface and surface joints were demonstrated by fabricating and testing complex, concept selection specimens. Cost of the baseline LFC aircraft was estimated and compared to the turbulent aircraft. The mission fuel weight was 21.7 percent lower for the LFC aircraft. Fuel costs were shown to be approximately $4,000,000 per year lower for the LFC aircraft. The calculation shows that the lower fuel costs for LFC offset the higher incremental costs of LFC in less than six months.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Laminar Flow Control (LFC) Fuel Conservation Wing Composite Structure Wing Chordwise Joint	[redacted] Subject Category 01

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	130	

AVAILABLE: NASA's Industrial Applications Centers