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A DEMONSTRATION PROGRAM PLAN
UTILIZING COMPOSITE REINFORCED METALS
FOR THE DC-8 HORIZONTAL STABILIZER STRUCTURE

Final Report

by

P. T. Sumida

June 1971

Prepared Under Contract No. NAS1-9953

for Langley Research Center

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

DOUGLAS AIRCRAFT COMPANY

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

This final report was prepared by McDonnell Douglas Corporation, Douglas Aircraft Company under NASA Contract NAS1-9953 and covers the work performed during the period of June 1970 to March 1971. The contract was administered under the direction of Richard A. Pride, Head, Composites Section, NASA Langley Research Center.

The individuals who made technical contributions to the program are listed together with their areas of activity.

P. T. Sumida	Program Manager
F. C. Allen	Structural Analysis
Dr. L. J. Hart-Smith	Structural Analysis
M. Ashizawa	Structural Design
H. W. Wilson	Structural Design
W. R. Johnson	Manufacturing Engineering
R. L. Zwart	Manufacturing Engineering
R. J. Palmer	Process Engineering
H. M. Toellner	Process Engineering
C. W. Matt	Operations Analysis
D. E. McCay	Testing
T. G. Weathermon	Testing

ABSTRACT

A feasibility study was performed and Demonstration Programs prepared to demonstrate the application of boron-epoxy reinforced 7075-T6 aluminum structure on the DC-8 aircraft. Preliminary design and feasibility studies were conducted on five candidate DC-8 structural components prior to selection of the horizontal stabilizer for detailed study. Weight savings of eight percent (based on the structural box) were estimated. The recommended Demonstration Program proposes four additional phases: (1) advanced development tests, (2) detail design and analysis, (3) fabrication of two flight and one test stabilizer, and (4) static proof, ground vibration, fatigue, and ultimate strength tests. In addition, a two year (minimum) flight service period for two horizontal stabilizers on airline DC-8s is planned. An alternate Demonstration Program proposes two additional phases culminating in the in-flight service experience of a single horizontal stabilizer on an airline DC-8 for approximately one year.

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A DEMONSTRATION PROGRAM PLAN
UTILIZING COMPOSITE REINFORCED METALS
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FINAL REPORT

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P T Sumida

SUMMARY

This report presents the results of a seven months study (Phase I) to evaluate the feasibility of composite reinforced metal (hybrid composite) structures on the DC-8 airplane. The study resulted in the preparation of Demonstration Program Plans to establish by flight service the feasibility of boron-epoxy reinforced 7075-T6 aluminum structure on the horizontal stabilizer of the DC-8. The plans cover the basic elements of design and analysis, materials, manufacturing, quality assurance, static and flight testing and in-flight service experience. The recommended plan, which is preferred by the NASA, Langley Research Center (LRC) and the Douglas Aircraft Company, consists of five phases including the Phase I study. The plan involves three horizontal stabilizers, i.e., two flight articles utilizing two DC-8 airplanes and a horizontal stabilizer for extensive static and fatigue ground testing (Figures A and B). An alternate plan, structured to be accomplished in two additional phases, resulted from a detailed investigation involving a single flight article.

The first six weeks of the Phase I study were directed to the selection of the structural component on the DC-8 airplane for further investigation. The NASA, LRC concurred with the Douglas recommendation of the horizontal stabilizer as the component for further study.

The DC-8 horizontal stabilizer provides a suitable means of demonstrating the feasibility of composite reinforced metals for primary aircraft structure.

1. The DC-8 is being flown by a large number of domestic and foreign carriers in a wide variety of configurations, i.e., all passenger to all freighter. It is reasonable to assume that many DC-8s will continue to serve the major airlines of the world for at least the time period of the proposed Demonstration Program.

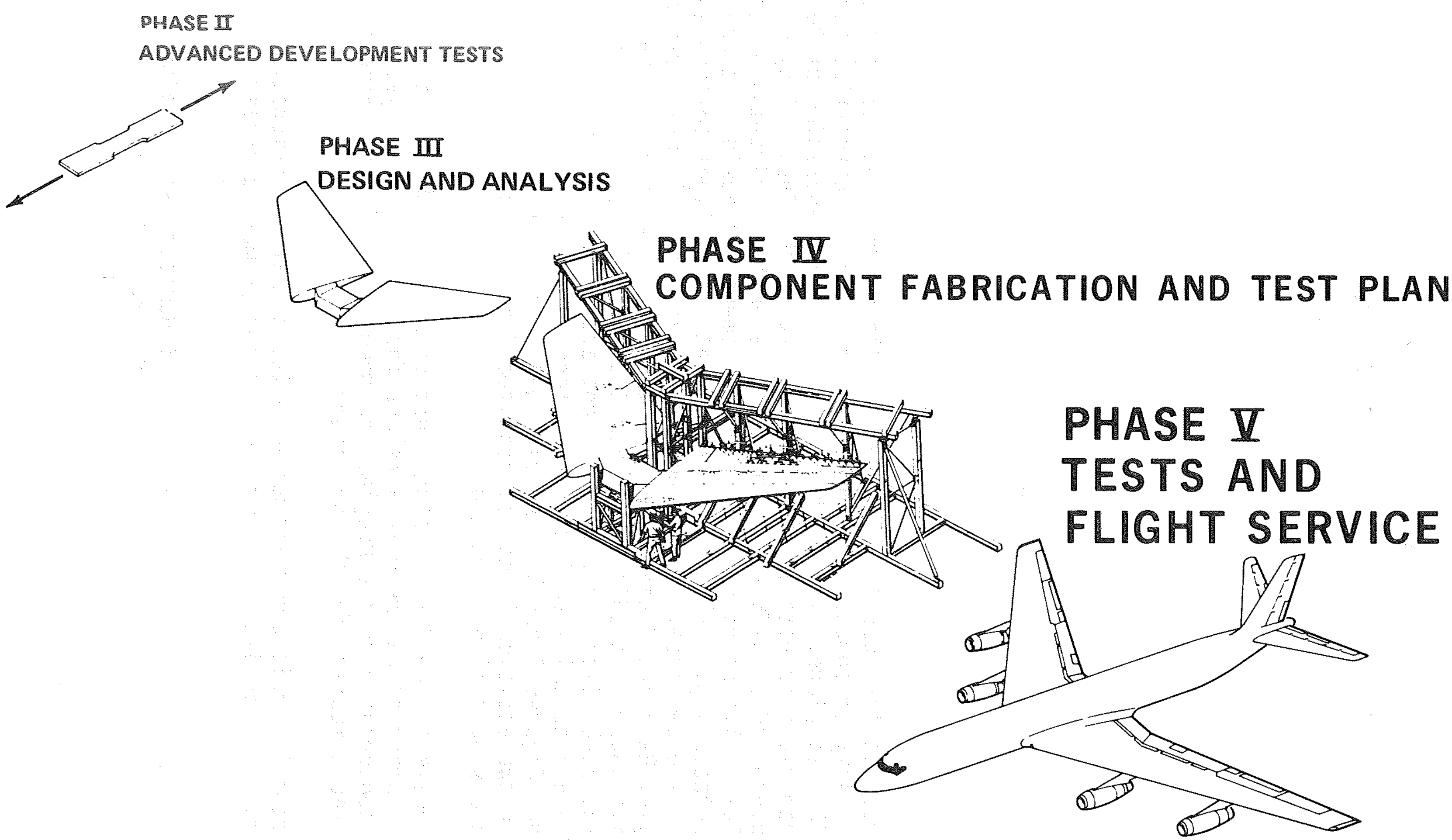

FIGURE A. DEMONSTRATION PROGRAM PLAN

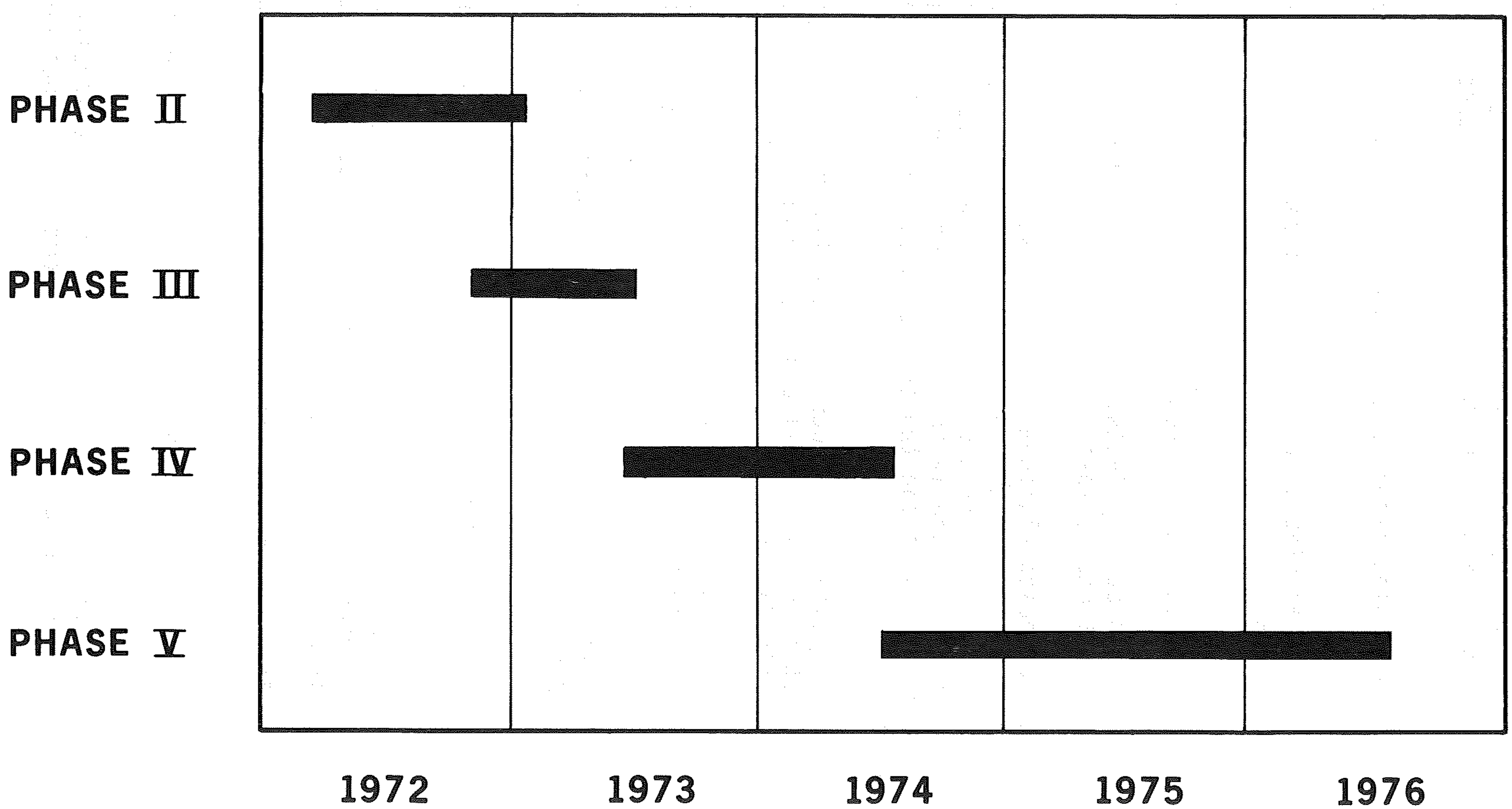

FIGURE B. PROGRAM SCHEDULE

2. United Air Lines (UAL) has indicated a willingness to participate in a flight service program with two DC-8s during the proposed 1974-1976 time period.

3. The DC-8 horizontal stabilizer is an economical and trouble-free component.

4. The present horizontal stabilizer is interchangeable among all models of the DC-8 airplane.

5. The stabilizer is basically an independently manufactured item which can be easily installed and/or removed with minimum aircraft down-time.

6. The development of necessary tests, specimen configurations, and critical load conditions is easily accomplished.

7. The existing broad based aluminum technology and the in-house understanding and capability for the design, analysis, fabrication, and assembly of all composite structures provides the necessary skills and experience for successfully integrating the unique and significant characteristics of each of the two technologies for application to hybrid composites.

8. Sophisticated analytical tools such as presently used computer programs are available within the McDonnell Douglas Corporation to support the design and analysis function.

The completion of the proposed Demonstration Program with the flight service of the DC-8 horizontal stabilizer could be advantageous for several reasons.

1. The reliability and actual weight savings of hybrid composite structures would be established.

2. Quality assurance procedures would be demonstrated as adequate for the hybrid composite structural configurations utilized on the DC-8 stabilizer.

3. Airline and FAA acceptance would be established for the utilization of composite reinforced structures on commercial jet transports in regularly scheduled airline service.

4. The near-term effectiveness of utilizing hybrid composite structures on large primary structure will be demonstrated.

5. Design allowables would have been established for hybrid composite structures.

6. Tooling and manufacturing capability would have been demonstrated for the fabrication and assembly of hybrid composite structures.

Work accomplished during the Phase I study included an evaluation of the many load carrying elements of the horizontal stabilizer for the potential application of composite reinforcement (Figures C and D). Upper limits on weight savings which include the effect of residual thermal stresses are defined for various area ratios of boron-epoxy reinforcement on aluminum. Some of the elements, such as spar caps and skin panels, offered substantial weight savings while other elements such as ribs and shear web stiffeners yielded insignificant savings in weight. Realistic weight savings of approximately 15 percent were calculated for the integrally stiffened skin panels. Weight savings of eight percent (based upon the total structural box) are estimated for the selective reinforcing of the skin panels and spar caps (Table i).

Considerable analytical effort was directed toward the area of bonded joints and the fatigue behavior of hybrid composites. The effect of thermally-induced locked-in residual stresses on the load transfer capacity of bonded joints was assessed for brittle and ductile adhesives. Thermal stress alleviation techniques such as bi-adhesive bonding (Douglas disclosure item) have been analytically developed. An analysis of cumulative fatigue damage due to gusts and ground-air-ground cycles was performed. The effect of changing the EI and GJ of the horizontal stabilizer with the change to hybrid composites was assessed from the standpoint of flutter speed margins and aerodynamic stability and control characteristics.

The techniques and problems associated with quality control and in-service inspection are outlined. The costs of composite prepregs are surveyed and documented. A basic manufacturing approach has been defined and involves the use of new metal parts for those structural members which are designed as hybrid composite members together with those metal parts which are to be redesigned to accommodate the hybrid composite parts.

The recommended Demonstration Program is outlined below:

Phase I, Feasibility Study and Program Plans:

This report describes the work accomplished during Phase I.

Phase II, Advanced Development Tests:

A review will be made of the current state-of-the art, several reinforcing structural concepts will be evaluated and the best concept selected. A test program will be conducted for (1) materials selection, (2) design allowables, and (3) substantiation and verification of the bonded joint representing the inboard end of the outboard structural assembly (Figure E).

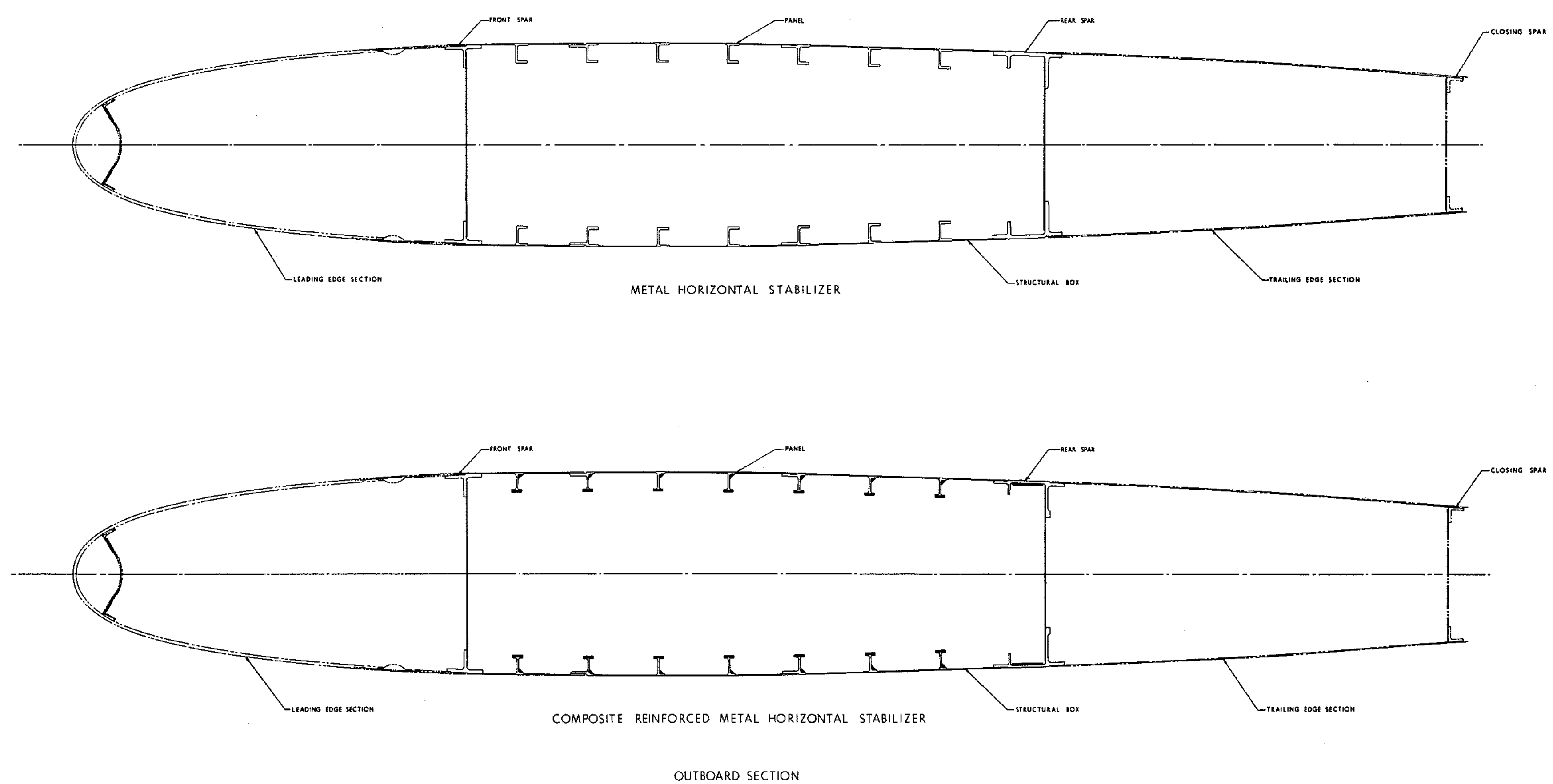

FIGURE C. COMPARISON OF ALL METAL AND HYBRID COMPOSITE HORIZONTAL STABILIZERS

TABLE i

SUMMARY OF WEIGHTS

ITEM \ WEIGHT		ALL METAL (lb)	HYBRID COMPOSITE (lb)	REDUCTION (lb)	REDUCTION (%)
CONSTANT SECTION CENTER BOX	SKIN PANELS	109.7	93.4	16.3	14.9
	FRONT SPAR: ASSEMBLIES CAPS (PART OF ASSY)	35.1 [15.8]	32.1 [12.8]	3.0 [3.0]	8.6 19.0
	REAR SPAR: ASSEMBLIES CAPS (PART OF ASSY)	48.8 [31.9]	42.8 [25.9]	6.0 [6.0]	12.3 18.8
	OTHER STRUCTURE	12.0	12.0	—	—
OUTBOARD STRUCTURAL BOX	SKIN PANELS	454.2	400.3	53.9	11.8
	REAR SPAR: ASSEMBLIES CAPS (PART OF ASSY)	209.5 [119.7]	190.3 [100.5]	19.2 [19.2]	9.2 16.0
	OTHER STRUCTURE	332.7	332.7	—	—
HORIZONTAL STABILIZER STRUCTURAL BOX		1202.0	1103.6	98.4	8.2

NOTE: [ ] DENOTES WEIGHTS WHICH ARE A PART OF SPAR ASSEMBLY WEIGHTS

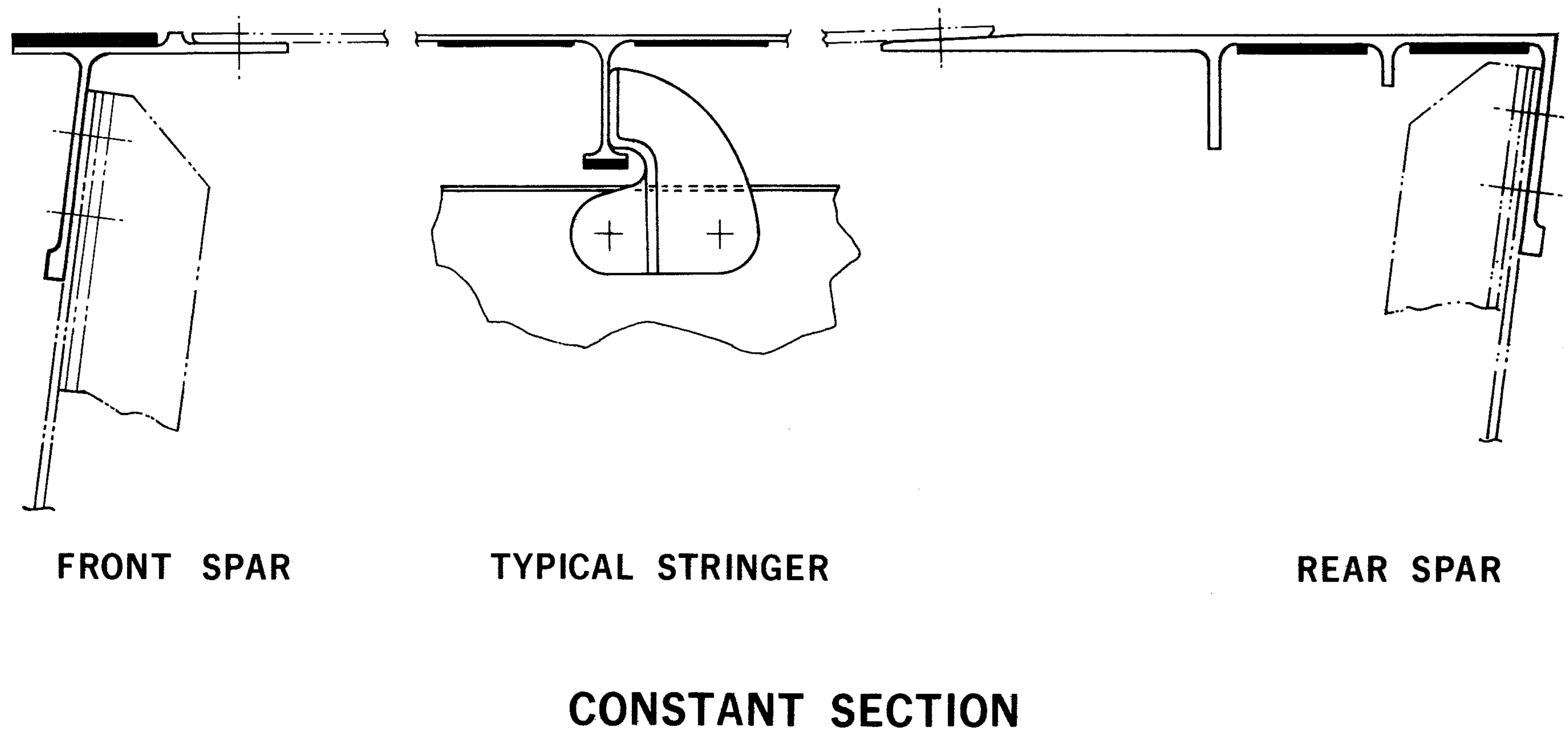

FIGURE D. HYBIRD COMPOSITE DC-8 HORIZONTAL STABILIZER

TABLE i

SUMMARY OF WEIGHTS

ITEM	WEIGHT	ALL METAL (lb)	HYBRID COMPOSITE (lb)	REDUCTION (lb)	REDUCTION (%)
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HORIZONTAL STABILIZER STRUCTURAL BOX		1202.0	1103.6	98.4	8.2

NOTE: [ ] DENOTES WEIGHTS WHICH ARE A PART OF SPAR ASSEMBLY WEIGHTS

- SELECT BEST STRUCTURAL CONCEPT FOR SKIN PANELS AND SPAR CAPS
- ESTABLISH CRITERIA AND LOADS
- DETERMINE ALLOWABLE STRENGTHS AND CONSTANTS
- PERFORM STRUCTURAL ANALYSIS
- PREPARE TEST PLAN
- FABRICATE COUPONS AND CONDUCT TESTS
- DESIGN AND TEST SUBCOMPONENT SPECIMEN OF COMPOSITE/METAL BONDED JOINT

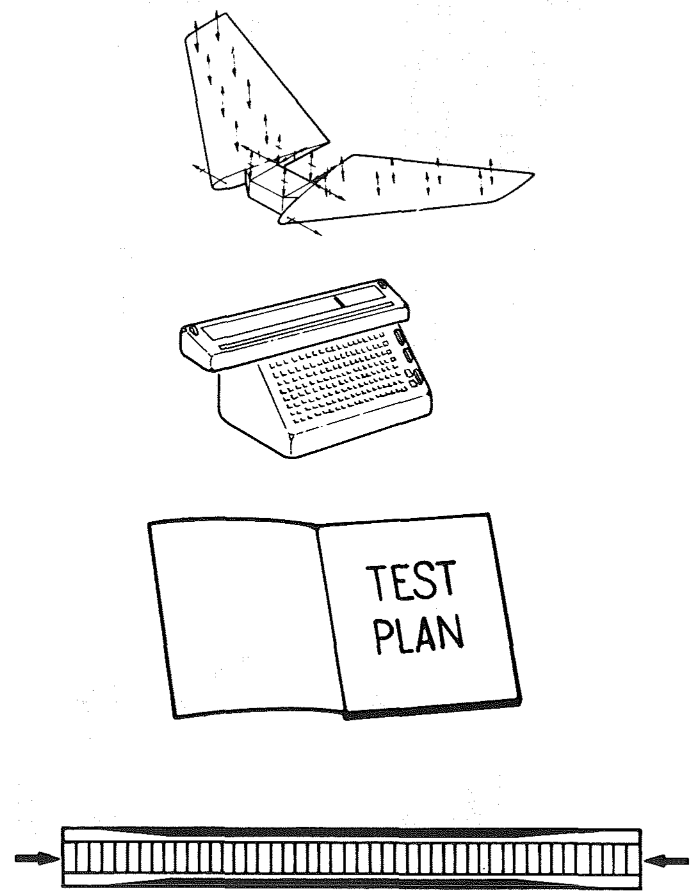


FIGURE E. PHASE II - ADVANCED DEVELOPMENT TESTS

Phase III, Detail Design and Analysis:

Detail design and analysis of the skin panels and spar caps for strength, stiffness, shear buckling, and column stability will be accomplished. A trade study and analysis will be conducted on scarfed and stepped joints. An analysis will be made of locked-in thermal stresses and their effect on the strength and fatigue life of the hybrid composite structural members. A subcomponent test program will be conducted to verify and substantiate the design of the panels and spar caps (Figure F).

Phase IV, Component Fabrication and Test Plan:

Tool design and fabrication and manufacturing processes will be completed. Three horizontal stabilizers will be assembled with one of the three stabilizers instrumented with flight flutter instrumentation. A test plan for static proof testing of three stabilizers and the fatigue and ultimate strength testing of one of the stabilizers will be completed and test drawings furnished (Figure G).

Phase V, Component Tests and Flight Service:

Each of the three stabilizers will be static tested to proof load (100% limit load). In addition, the ground test article will be tested for fatigue life and ultimate strength. Two of the composite reinforced horizontal stabilizers will be shipped to UAL's maintenance base in San Francisco and installed on DC-8 airplanes. The all metal horizontal stabilizers will be held in storage at the airlines maintenance facility for the duration of the flight service period. One of the installed stabilizers will be given a ground vibration test. Flight acceptance checks will be required for both airplanes with one of the airplanes participating in a flight certification period. After FAA certification, the airplanes will be returned to the domestic airline for a flight service period of two years minimum. Finally, the airplanes will be refurbished with the original horizontal stabilizers (Figure H).

- COMPLETE ALL ENGINEERING DRAWINGS
- CALCULATE WEIGHT REDUCTION FOR HORIZONTAL STABILIZER
- CALCULATE NET WEIGHT AND BALANCE CHANGE FOR DC-8
- PERFORM STRUCTURAL ANALYSIS
- PERFORM AEROELASTIC AND FLUTTER ANALYSIS
- ESTABLISH RELIABILITY AND QUALITY ASSURANCE PROCEDURES
- PREPARE RELIABILITY AND QUALITY ASSURANCE DOCUMENTS AND CONDUCT TESTS AS NECESSARY

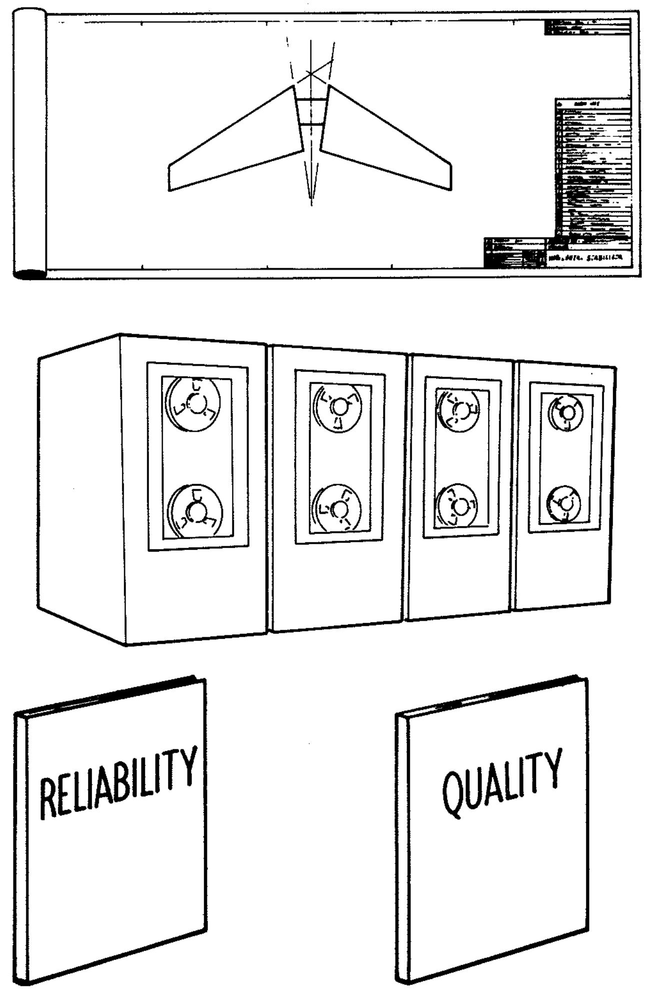
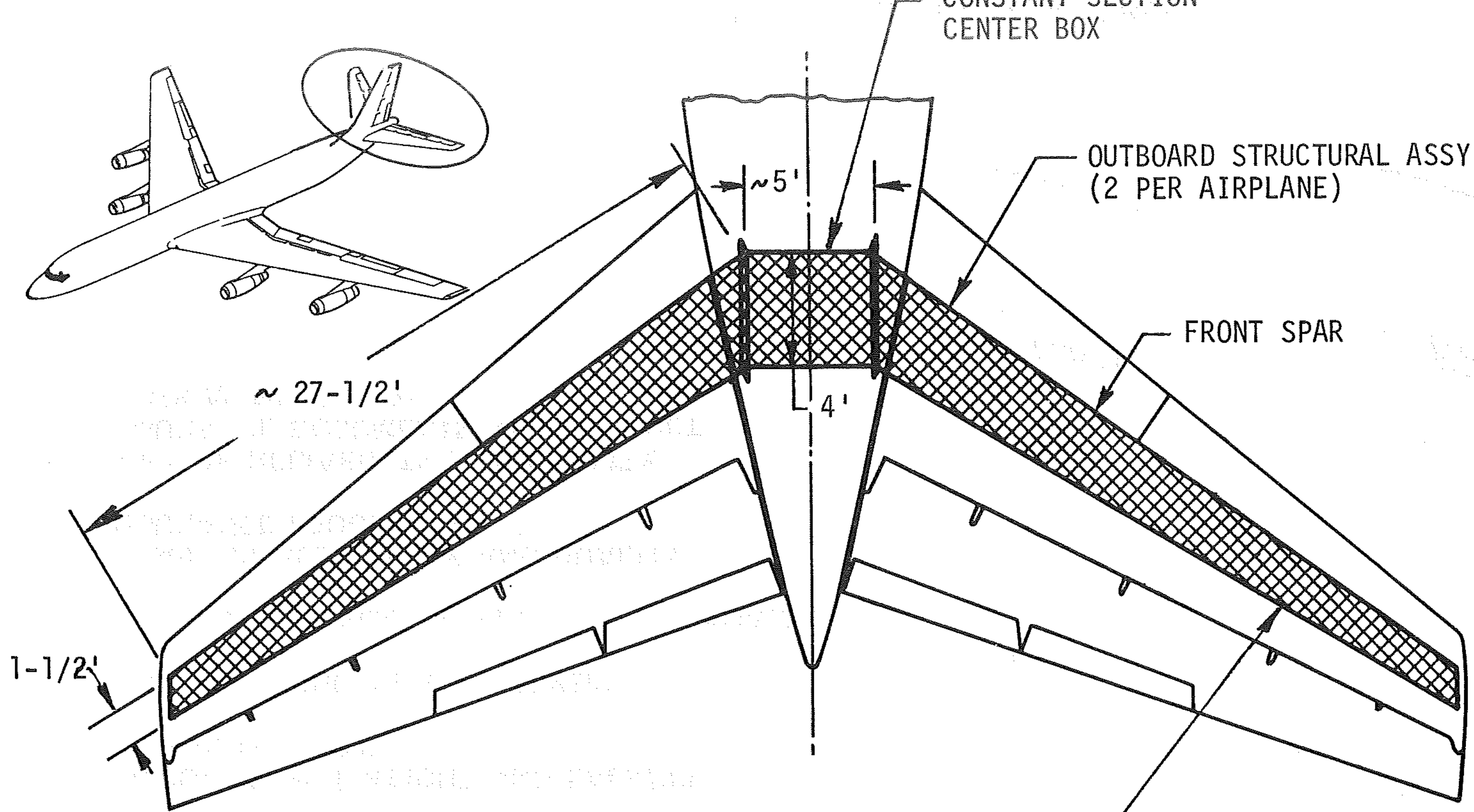


FIGURE F. PHASE III - DETAIL DESIGN AND ANALYSIS

1. COMPONENT FABRICATION



2. TEST PLAN

- STATIC TESTS
- FATIGUE TEST
- ULTIMATE STRENGTH TEST

FIGURE G. PHASE IV - COMPONENT FABRICATION AND TEST PLAN

1. STATIC (PROOF) TESTS

   THREE HORIZONTAL STABILIZERS (2 FLIGHT AND 1 GROUND TEST ARTICLE)

2. FATIGUE TEST

   GROUND TEST ARTICLE

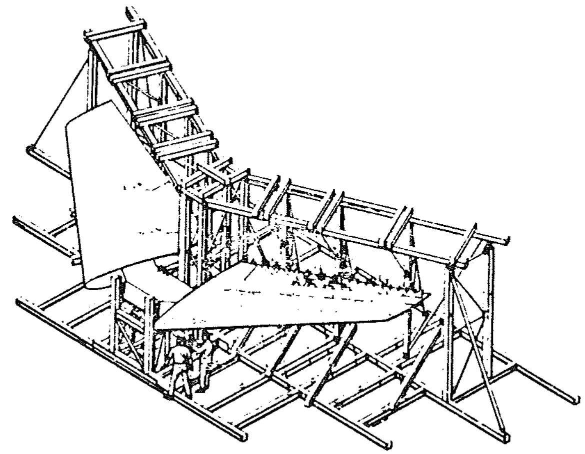
3. ULTIMATE STRENGTH TEST

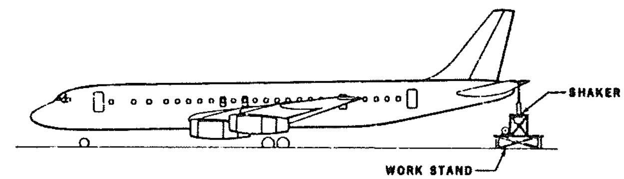
   GROUND TEST ARTICLE

4. GROUND VIBRATION TEST

5. FLIGHT TEST

6. FLIGHT SERVICE





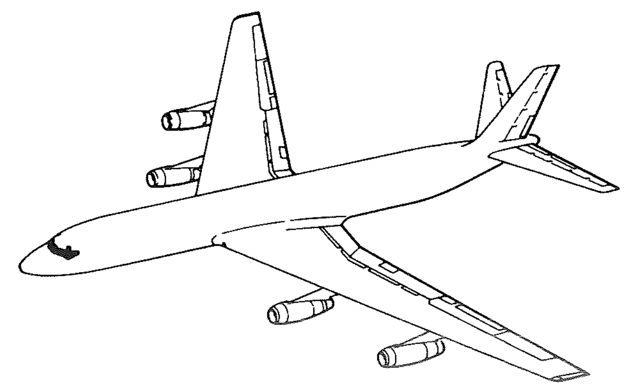


FIGURE H. PHASE V - COMPONENT TESTS AND FLIGHT SERVICE

SECTION 1

INTRODUCTION

The development of hybrid composites has been led largely by the NASA, LRC, both by in-house studies and by contracts to the aerospace industry. The motivation behind the concept is aptly expressed in the original paper by Zender and Dexter (1): "The concept of bonding high performance filaments to metal structures builds upon the large existing background of fabrication technology for aerospace structures. This important practical advantage along with the potential weight saving indicated suggest the desirability of a rapid development of this concept in the aerospace industry." The high values of specific strength and modulus of many filamentary materials have motivated substantial efforts to utilize them in aerospace structures to save weight. Considerable progress has been made in all-composite structures but particularly in regard to mechanical joints in these anisotropic materials, much development work remains to be done before their full potential can be realized in regular production. The philosophy underlying hybrid composites is to design the joints in metal alone and to use the filamentary composites for uniaxial loading, in which they are most efficient. By using parallel metal and filamentary composite load paths, a degree of failsafe capability is attained. It is expected that the use of hybrid composites rather than pure composite structures will accelerate the introduction of advanced filamentary composites into commercial aircraft structures but, while this concept eliminates certain problems inherent in all-composite structures, it introduces new ones in their place - problems stemming directly from the thermal expansion mismatch of the constituents of the hybrid composites.

It is appropriate at this point to review the considerable work already undertaken on hybrid composites. At the NASA, LRC, Zender, Dexter and Davis (1, 2 and 3) have performed extensive theoretical and experimental investigations, largely on axially reinforced tubes. At Avco, Henshaw, Roy and Russell (4) have done considerable development work with the concept of infiltrating boron-epoxy into hollow elements. At LTV, McQueen, McClaren and Martin (5) investigated a "new stiffened panel concept - integrally formed structures" made by bonding together two thin sheets of metal, one of which has the stiffeners formed in it. Their hybrid composite version of this structure encloses uniaxial filamentary composite between the sheets. At Boeing, Lager and June (6) designed and fabricated a hybrid composite aircraft floor beam and demonstrated a substantial weight reduction in comparison with the comparable aluminum floor beam of the Boeing 707. Also at Boeing, Stensrud and Fosha, Jr (7) examined the application of hybrid composite stiffeners to the fuselage of their SST. Working under NASA contract NAS1-8858, Oken and June (The Boeing Company) have investigated several of the critical details of hybrid composite construction, in particular, stepped-lap metal attachments bonded to the ends of the filamentary composites. They have conducted both theoretical and experimental investigations of bonding development, residual thermal stresses, composite load transfer, compression stress-strain, plate bending, plate buckling, column crippling and buckling, and sandwich crippling and buckling, and tested a number of concept verification panels. Also working under a NASA contract NAS1-9540, at the Lockheed-Georgia Company, Petit (8) has made an application study of spanwise filamentary composite reinforcement on the C-130 wing center box. The study was aimed at improving the wing center box fatigue life. As a result of the study, additional reinforcing was required over that for static strength alone. Wing center box weight savings of 13 percent were estimated with boron-epoxy reinforcement and 9 percent with graphite-epoxy reinforcement.

Recent, as yet unpublished work by Illg at the NASA, LRC has concentrated on fatigue tests. These have confirmed the existence of problems arising from locked-in stresses induced by the thermal mismatch of the hybrid constituents. Boeing also has conducted fatigue tests (as yet unpublished) under the NASA, LRC Phase II contract NAS1-8858 on hybrid composite specimens. Independent test programs at Douglas and the NASA, LRC, have demonstrated the effectiveness of bonded composite crack-stoppers.

As a follow-on to the preceding investigations, Douglas Aircraft Company has undertaken a feasibility study investigating the application of hybrid composite structure to a major structural component of the DC-8 aircraft. This report covers the results of this study which involved component selection, design application of hybrid composites to the selected component and a proposed Demonstration Program Plan.

SYMBOLS

NOTATIONS

A	Area (in^2)
E	Modulus of elasticity (lb/in^2)
M_D	Dive mach number
S	Surface area (in^2)
T	Temperature, Fahrenheit
ΔT	Change in temperature, Fahrenheit
V_D	Design dive speed (knots)
V_F	Flutter Speed (knots)
V_{mo}	Maximum operating speed (knots)
k	Volume fraction of metal
x,y,z	Cartesian coordinate
α	Coefficient of thermal expansion ($in/in/^{o}F$)
γ	Shear strain (in/in)
ε	Strain (in/in)
μ	Poisson's ratio
ρ	Density (lb/in^3)
σ	Normal stress (lb/in^2)
τ	Shear stress (lb/in^2)
$\emptyset$	Scarf angle

Subscripts

Al Aluminum

B Boron

C Composite, filamentary

H Hybrid

M Metal

Me Metal equivalent

R Residual

Ti Titanium

Abbreviations

Al Aluminum

Comp Compression

Hor Horizontal

Mag Magnesium

Mod Modulus

Reinf Reinforcement

Stl Steel

Str Strength

3M Minnesota Mining and Manufacturing Company

Ti Titanium

Ult Ultimate

Vol Volume

SECTION 2

HORIZONTAL STABILIZER PRELIMINARY DESIGN AND ANALYSIS AND FABRICATION METHODS

2.1 DESIGN AND ANALYSIS

Five candidate components of the DC-8 were evaluated on the basis of program feasibility resulting in the selection and approval by NASA, LRC, of the horizontal stabilizer for further study. Details on the selection of the horizontal stabilizer are given in Appendix A.

2.1.1 Description of All Metal Horizontal Stabilizer

The horizontal stabilizer is a continuous unit and is attached to the airplane fuselage by two pivot points at the rear spar and by two screw jacks at the front spar. The structural torque box and spars pass uninterrupted through the fuselage to eliminate any fatigue-sensitive discontinuities (Figure 1). Manufacturing and assembly breaks are provided at the sides of the fuselage. The integrally stiffened skin panels of the outboard sections and the center section are machined into continuous chordwise integral fittings at these locations and are fastened together by closely-spaced tension bolts. The operating bulkheads are also installed with the same bolts.

The outboard section of the horizontal stabilizer consists of a single primary torque box composed of front and rear spars plus two secondary torque boxes formed by the leading edge and trailing edge structure with the closing spar. The upper skin varies in thickness from 0.097 inch at the root to 0.045 inch at one-third semi-span and is constant outboard. The lower skin is similar but with an 0.131 inch thick root section. The minimum thickness is set by manufacturing requirements. There are eight stringers at the root spaced between spars and running parallel to the rear spar. The stringers scarf off prior to intersecting the front spar or reaching their outboard extremity. Rib spacing in the primary box is 24 inches. Most of the shear substructure is of minimum gage and incorporates a large number of flanged lightening holes.

The center box section which is inside the fuselage is of constant chord. It utilizes the same integrally stiffened skin and stringer structure as the outboard primary box. The top skin is 0.076 inch thick and the bottom skin is 0.093 inch thick.

All major structure on the horizontal stabilizer is of 7075 aluminum alloy. The integrally-stiffened skin planks are divided into three detail parts by spanwise seams to provide failsafe capability.

2.1.2 Design Philosophy for Hybrid Composite Horizontal Stabilizer

The basic underlying philosophy for the present composite reinforced metal concept is the use of filamentary composites in small amounts and in judicious locations to provide significant weight savings with minimum risk and low cost. An appreciation of the basic guidelines which have evolved for implementing the composite reinforced concept is needed to understand the reason for the structural configurations to be discussed.

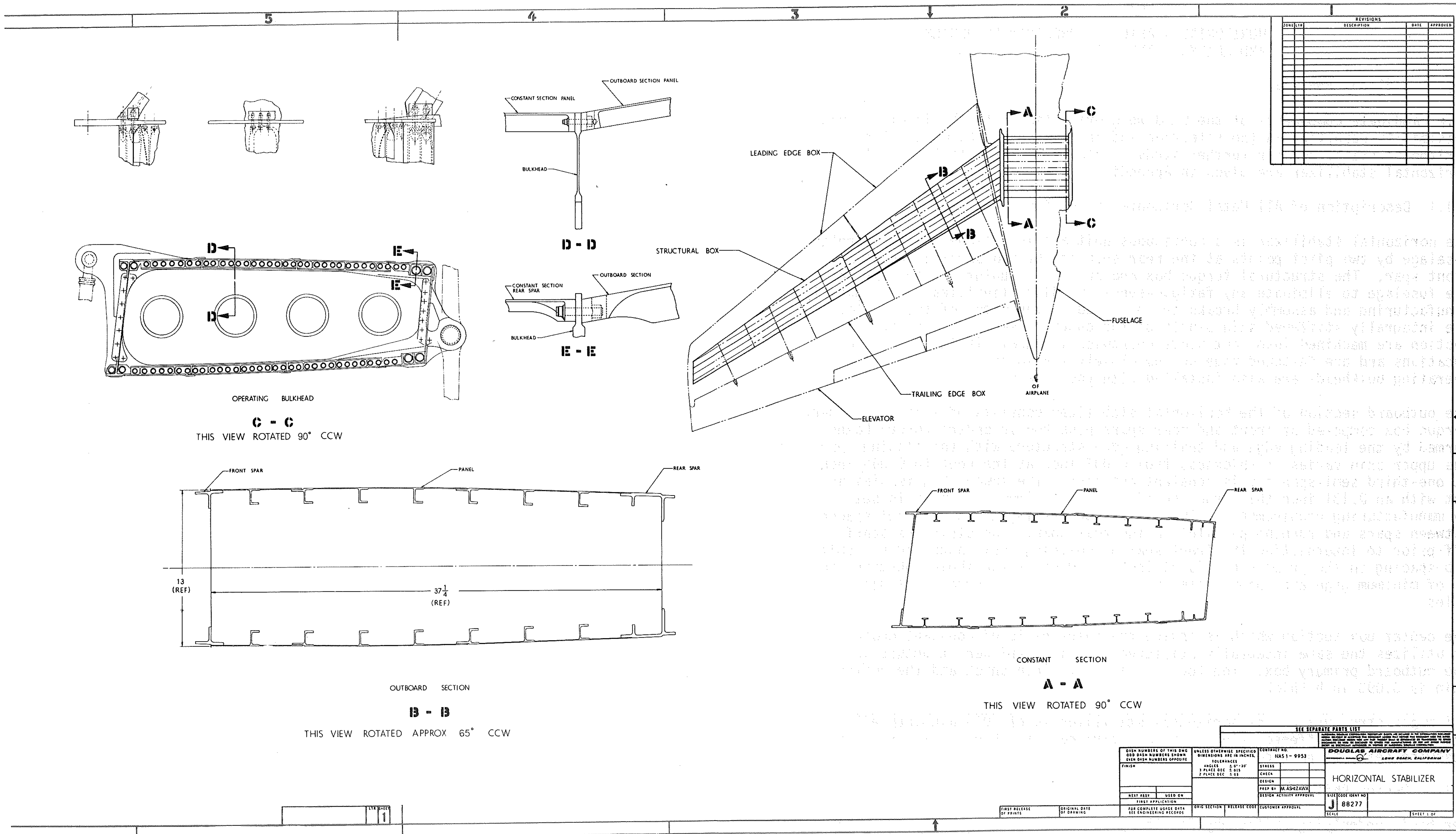


FIGURE 1. ALL METAL DC-8 HORIZONTAL STABILIZER

- o Each element of the structural assembly must be either composite reinforced metal or 100 percent metal

- o Existing all metal designs which are redesigned for composite reinforced metal must retain the basic structural capability as the all metal design, e.g., the failsafe characteristics of any all metal design must be retained in the composite reinforced design

Several additional design constraints were imposed by Douglas program management on the design of the hybrid composite components.

- o Loft lines (aerodynamic shape) and tooling and manufacturing planes, e.g., front and rear spars, are not to be altered

- o General location and configuration of mechanical joints are not to be changed, or if necessary, change should be small

- o Hybrid composite components should not require tooling and manufacturing development, i.e., capabilities beyond present understanding and experience

- o Reinforcement is limited to unidirectional filamentary composites for carrying uniaxial loads

Prior Douglas IRAD analysis revealed that hybrid composite structures can be efficient under unidirectional loading, but not under shear. Therefore, the philosophy employed in designing this hybrid component has been one of using metal alone for shear loads and concentrating the composite reinforcement into the stiffeners and spar caps.

2.1.3 Structural Design

The entire horizontal stabilizer structure was evaluated for the incorporation of hybrid composite parts with the basic objective of utilizing boron-epoxy for maximum weight savings while maintaining reasonable program costs. Many of the structural members or assemblies were minimum gage material or considered too complex and impractical for the application of hybrid composites and were not considered for redesign. Several structural assemblies and detail parts however appeared to have potential for a meaningful weight savings and were considered as potential hybrid composite parts. Design studies showed that selective reinforcing with boron-epoxy of the integrally stiffened skin panels and spar caps would yield a weight saving of approximately 8 percent (based on the structural box). This discussion on weight savings is in Appendix B.

The following components are considered for the application of hybrid composites:

a. Constant Section Center Box

- o Integrally stiffened skin panels

- o Front Spar caps
- o Rear Spar caps

b. Outboard Structural Box Section

- o Integrally Stiffened skin panels
- o Rear Spar caps

Preliminary designs for the skins and spar caps for the constant section center box and the outboard structural box are discussed in the following paragraphs.

Constant Section Center Box

Skin Panels

The constant section of the stabilizer composed of six 7075-T6 Al alloy plate panels with integrally machined stiffeners and end fittings is amenable to the hybrid composite design concept.

A typical hybrid composite panel is illustrated in Figure 2. The thickness of the skin is reduced to .045 (minimum gage requirement) from the original metal skin thickness of .076. The loss of EA due to the thickness reduction is compensated by bonding the unidirectional boron composite having the same effective EA to the skin.

The stringer webs and flange thicknesses are reduced to effect a weight savings while maintaining the capability of carrying the shear load. The flange width is narrowed down to one-half inch symmetrically with respect to the centerline of the stringer. The undirectional boron is bonded to the top of the flange maintaining the same bending stiffness, EI, for each stiffener plus effective skin as for the metal structure.

The composite reinforcements terminate before reaching the end fittings through a one degree taper scarf joint as shown in Section B-B, Figure 2.

The hybrid composite panels can be machined from existing metal panels. Redesigning of the shear clips is not required.

Front Spar Caps

The redesigned upper and the lower spar caps are similar in construction having a "T" cross-section. The leg and flange of the spar cap are reduced to the thickness needed to transfer fastener loads. Added rectangular flanges at the end of the leg and at the middle of the flange increase buckling (crippling) strength. Since one flange of the spar cap has no fastener holes, the width has been considerably reduced on that side (Figure 3).

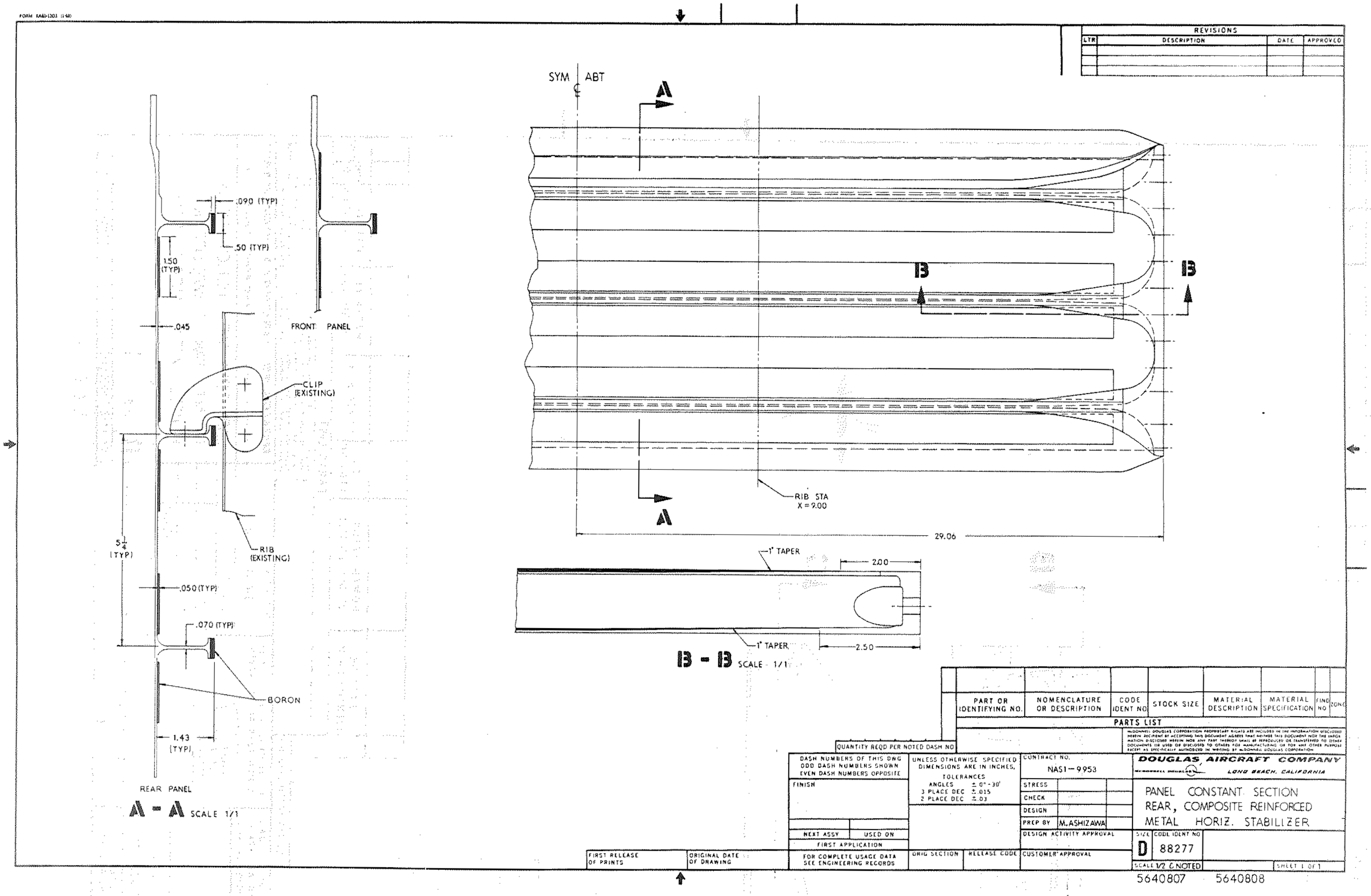


FIGURE 2. PANEL - CONSTANT SECTION REAR, COMPOSITE REINFORCED METAL

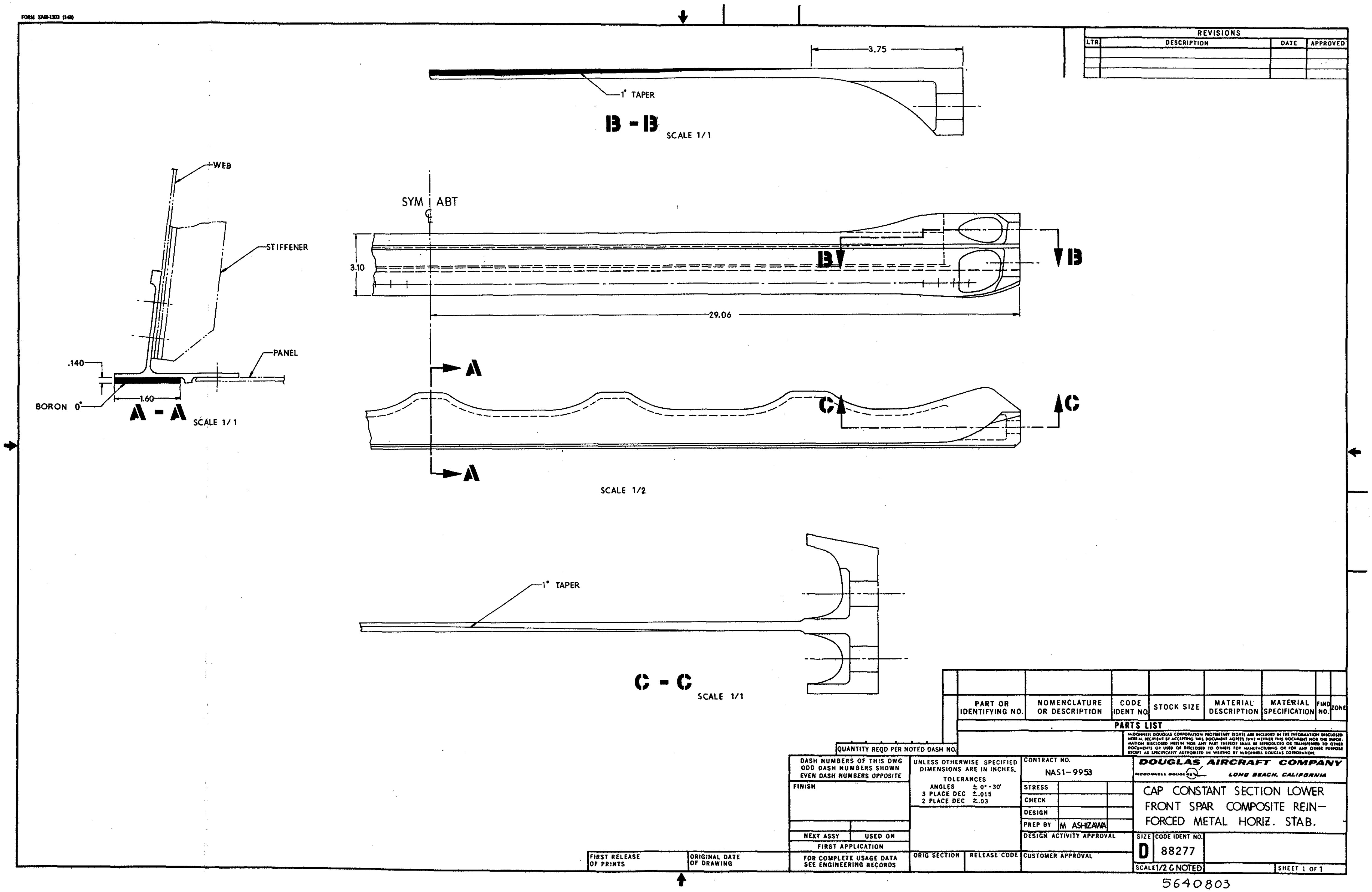


FIGURE 3.

CAP-CONSTANT SECTION LOWER FRONT SPAR, COMPOSITE REINFORCED METAL

Unidirectional boron composite is bonded to the exterior outer surface of the flange. This configuration has several advantages: (1) a single wide composite piece can be bonded to the metal rather than two narrow pieces, (2) there is less possibility of damage during assembly to the shear web structure, and (3) inspection and testing are made easier.

Again, the integral end fitting area is untouched. Transferring the load in the composite to the spar cap is done by the one degree taper scarf joint. Reduced flange width is restored near the end fitting by a gradual slope so that discontinuity in geometry is at a minimum.

Rear Spar Caps

The rear spar design concept is similar to that of the front spar cap. The two unidirectional boron composite strips bonded to the inside of the spar cap are stopped short of the drain holes (lower surface only) to minimize the stress concentration (Figure 4).

Outboard Structural Assembly

Skin Panels

The upper and lower exterior skins are an assembly of three integrally stiffened skin planks similar in detail design and structural configuration. The hybrid composite design of one panel is representative of each of the other panels (Figure 5). The panel is machined from 7075-T6 aluminum plate. The plate has machined integral stiffeners and stringer end fittings similar to the all metal design. To minimize the rotation of the integral stringers, the stringers are symmetrical "T"s with respect to the vertical axis as shown in Figure 5. Unidirectional boron composite (0.090 thick) is bonded to the top of the stringer flange, with one degree taper joints at both ends for an efficient load transfer. The integral stringers are supported by shear clips at the rib stations spaced 24 inches apart. At the intersection of the stringer and skin panel, triangular shaped unidirectional boron composite is placed such that it does not interfere with the shear clips (Figure 6). The clips provide the necessary load transfer capability between the skins and provide rotational fixity for the stringers.

The hinge rib shear clip is modified to accommodate the triangular shape composite but the shear clips at the other ribs did not require modification.

The skin thickness of the panel is 0.045 inch except at the root end and at the hinge rib stations where it is 0.070 inch thick. The minimum thickness of 0.045 inch is required for two reasons: (1) to prevent the skins from buckling during machining and to make it possible to have reasonable tolerances on skin thickness, and (2) to provide an adequate electrical current path to sustain lightning strikes without damage.

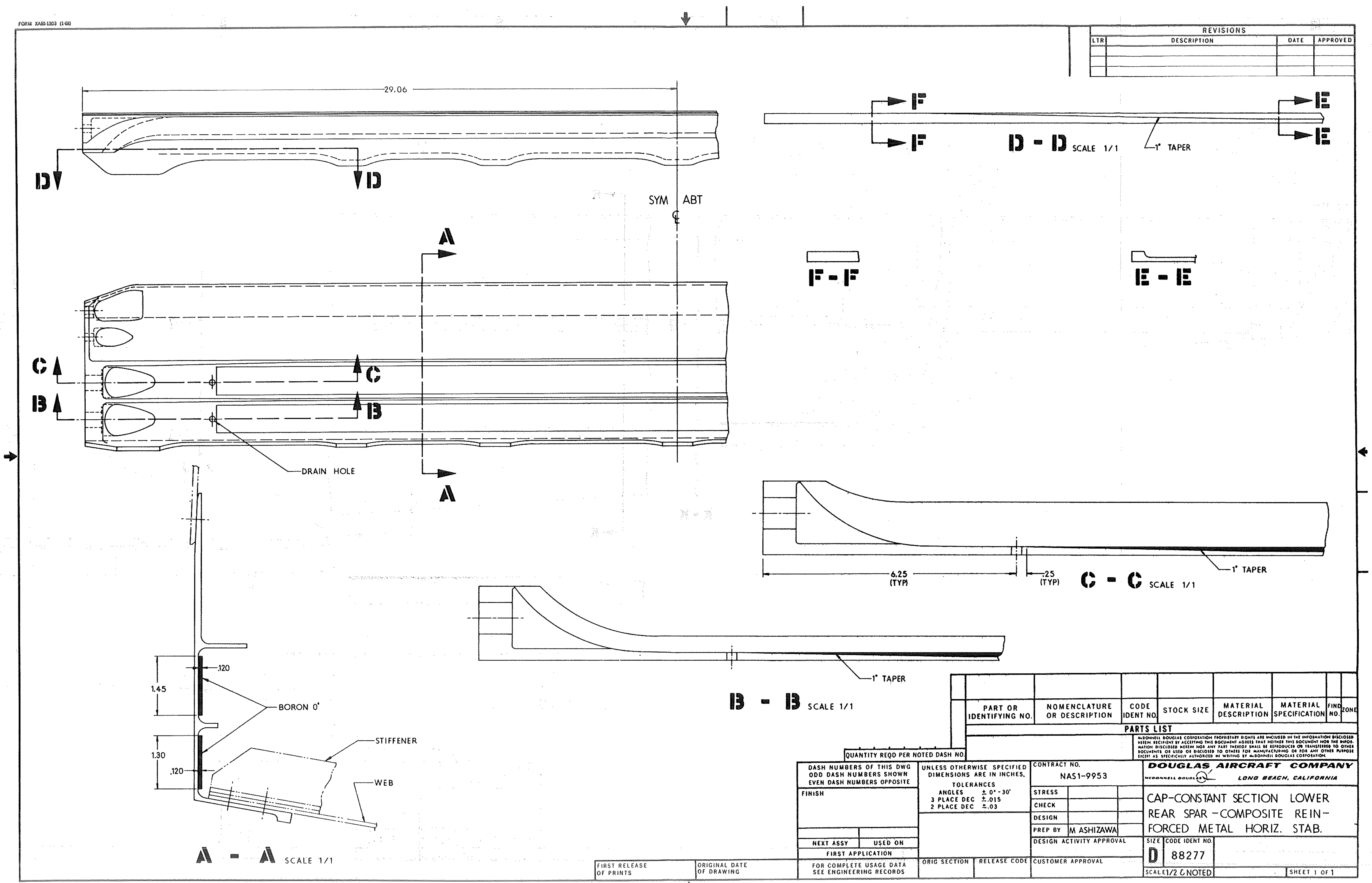


FIGURE 4.

CAP-CONSTANT SECTION LOWER REAR SPAR, COMPOSITE REINFORCED METAL

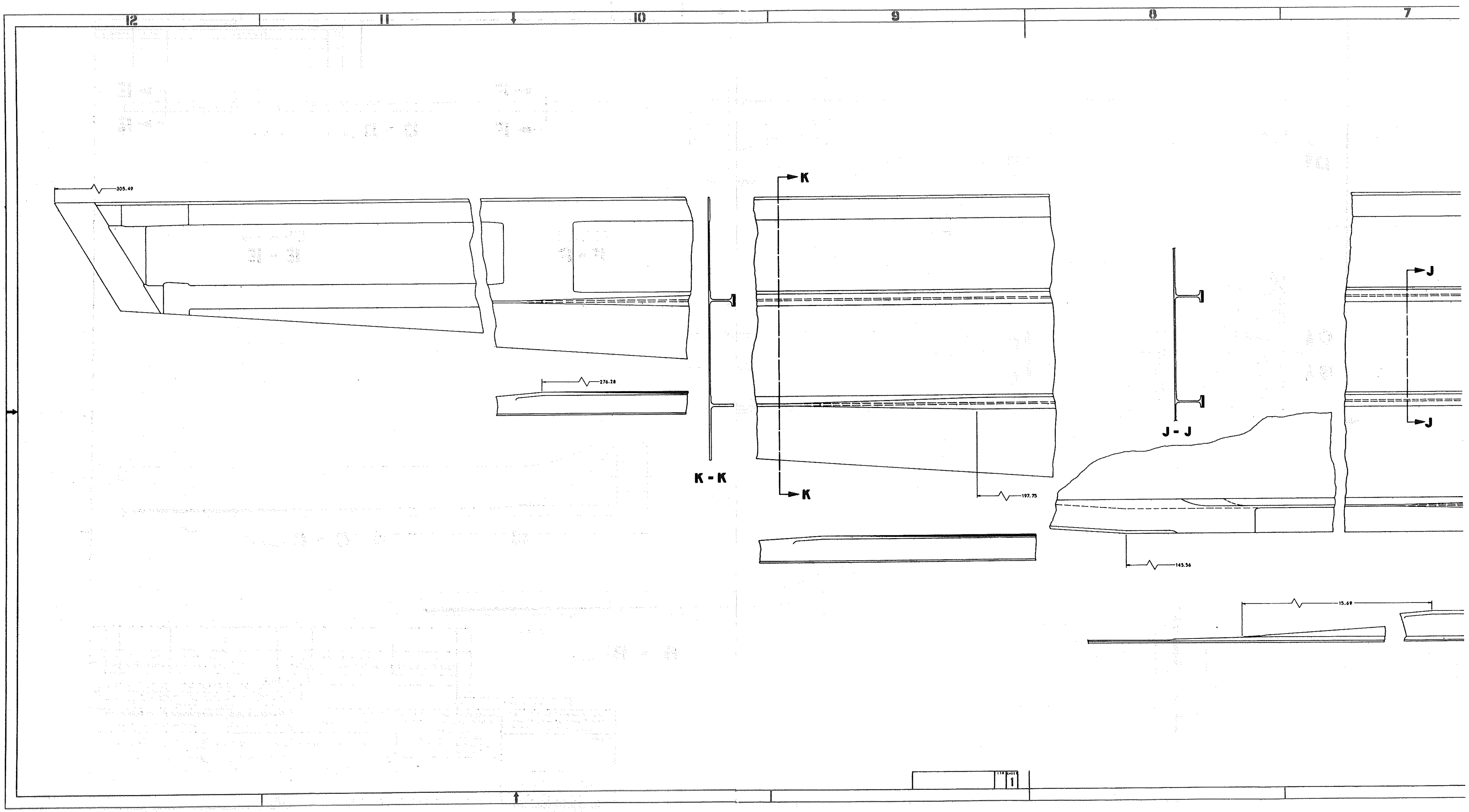


FIGURE 5. PANEL-OUTBOARD SECTION SKIN AND STRINGER, COMPOSITE RE-INFORCED METAL

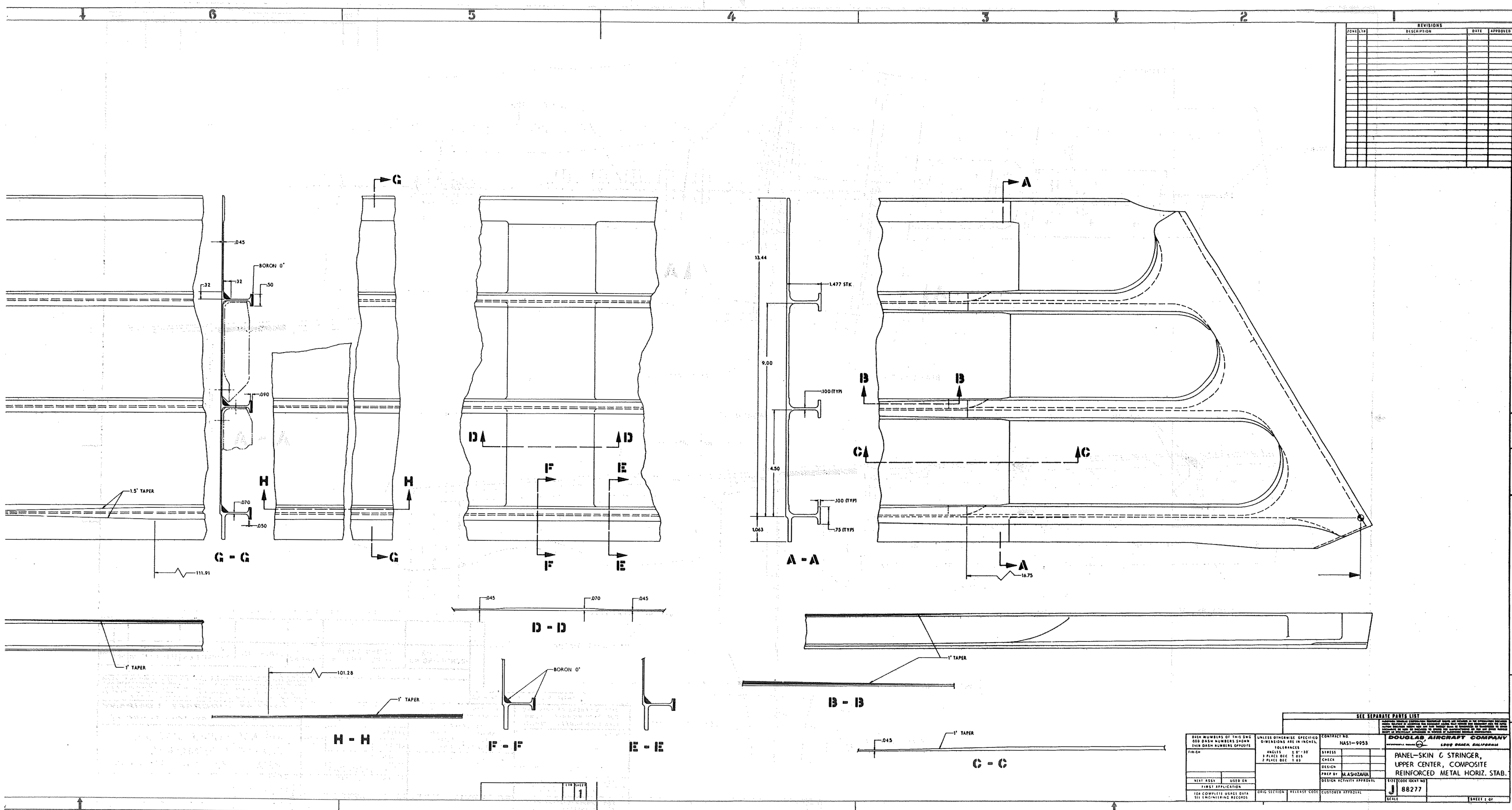


FIGURE 5. PANEL-OUTBOARD SECTION SKIN AND STRINGER, COMPOSITE REINFORCED METAL (CONTINUED).

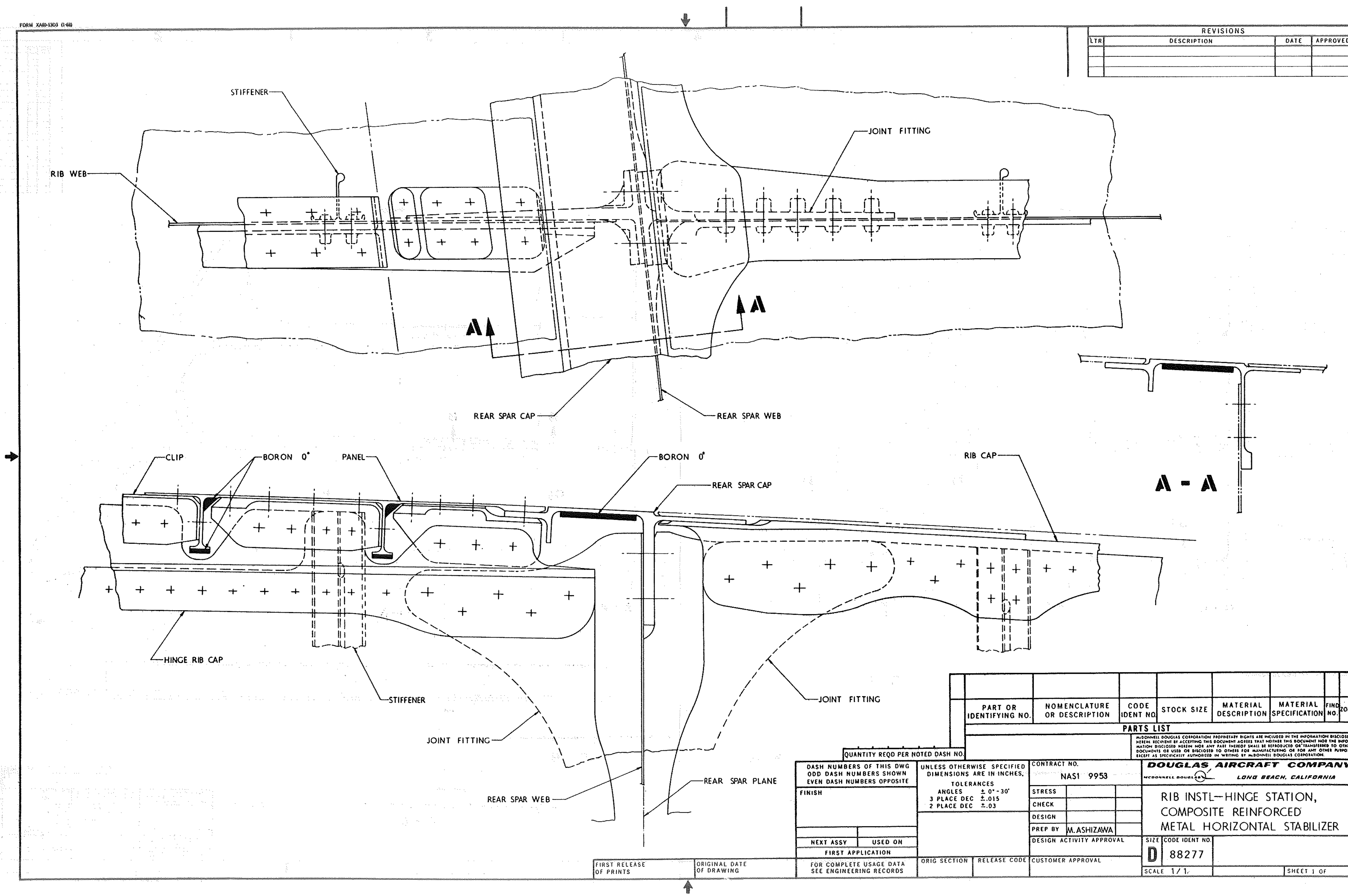


FIGURE 6. RIB INSTALLATION-HINGE STATION, COMPOSITE REINFORCED METAL

Rear Spar Caps

The design of the upper and the lower spar cap is similar. The lower rear cap is shown in Figure 7. The unidirectional boron composite strip which is two inches wide at the inboard end and tapers down to one-half inch at the outboard end (Figure 7) is bonded to the 7075-T6 modified aluminum spar cap. The thickness of the composite strip also varies linearly from 0.140 inch at the inboard end to 0.100 inch at the outboard end. The length of the composite strip is approximately one-third of the total length of the spar cap. The outer two-thirds of the cap offers such an insignificant weight savings that it remains an all metal section. One degree taper joints are used at both ends of the strip for load transfer. The redesigned shorter vertical leg which is required for one-quarter of the total length (semi-span) at the inboard end maintains the same b/t as the all metal design. The longer vertical leg (all metal) of the spar cap is redesigned to the reduced thickness while maintaining its ability to resist the fastener loads. A new flange is added to the free end of the vertical leg to increase buckling strength. The reduced thickness and the added flange are uniformly changed back to the original shape at every hinge rib location to accommodate the existing hinge rib joint fittings (Figure 6).

The detail design modifications depend on the failure mode of the element under consideration. The existing spar caps are critical in local crippling while the stringers of the skin panels are critical in column buckling (subject to a reduction because of the shear loads in the skin). As a preliminary design guide, therefore, the hybrid stringers were so proportioned that the stringer and adjacent effective skin maintained the same bending stiffness EI as for the existing metal structure. The further constraint of minimizing the changes in the overall bending stiffness of the stabilizer dictated that the same effective EA be maintained for the stiffened skin combination. There are two reasons for this latter constraint. The first is to avoid a major redistribution of elevator loads, thereby eliminating the need for reanalyzing, and perhaps redesigning, the elevator. The second is to minimize the changes associated with the flutter characteristics.

An attempt was made to reduce substantially the weight of the spar caps by reducing the metal thickness to that needed to transfer fastener loads, using spanwise boron-epoxy (bonded directly to the metal) to maintain the EA value for the cross-section. Transverse plies of graphite-epoxy were to be added, concentrated on the other side of the boron-epoxy to restore the crippling resistance. The choice of graphite fibers was dictated by small radii bends needed in the transverse plies. In order to eliminate the problem of drilling holes through boron, a strip of glass-epoxy was proposed to be substituted in the layup for such areas. This concept was discarded for two reasons. IRAD tests made at Douglas on a hybrid channel simulating the spar cap identified a plane of weakness at the boron-to-graphite interface. Early fatigue tests performed at the NASA, LRC, with a low temperature curing adhesive revealed that debonds were propagated from around stress concentrations. Subsequent tests using an elevated temperature curing adhesive indicated no problems exist in terms of debonds.

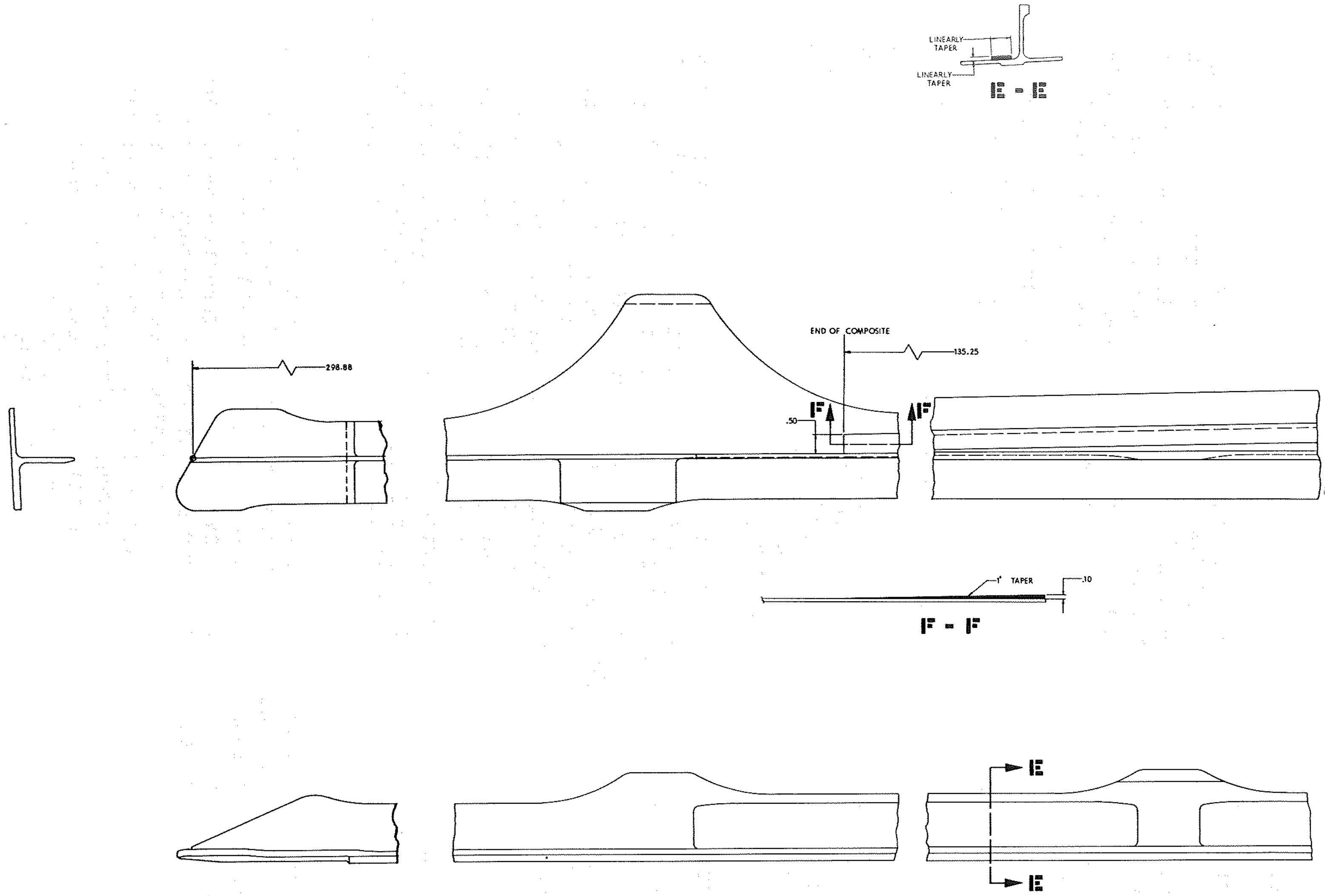


FIGURE 7. CAP-OUTBOARD SECTION LOWER REAR SPAR, COMPOSITE REINFORCED METAL

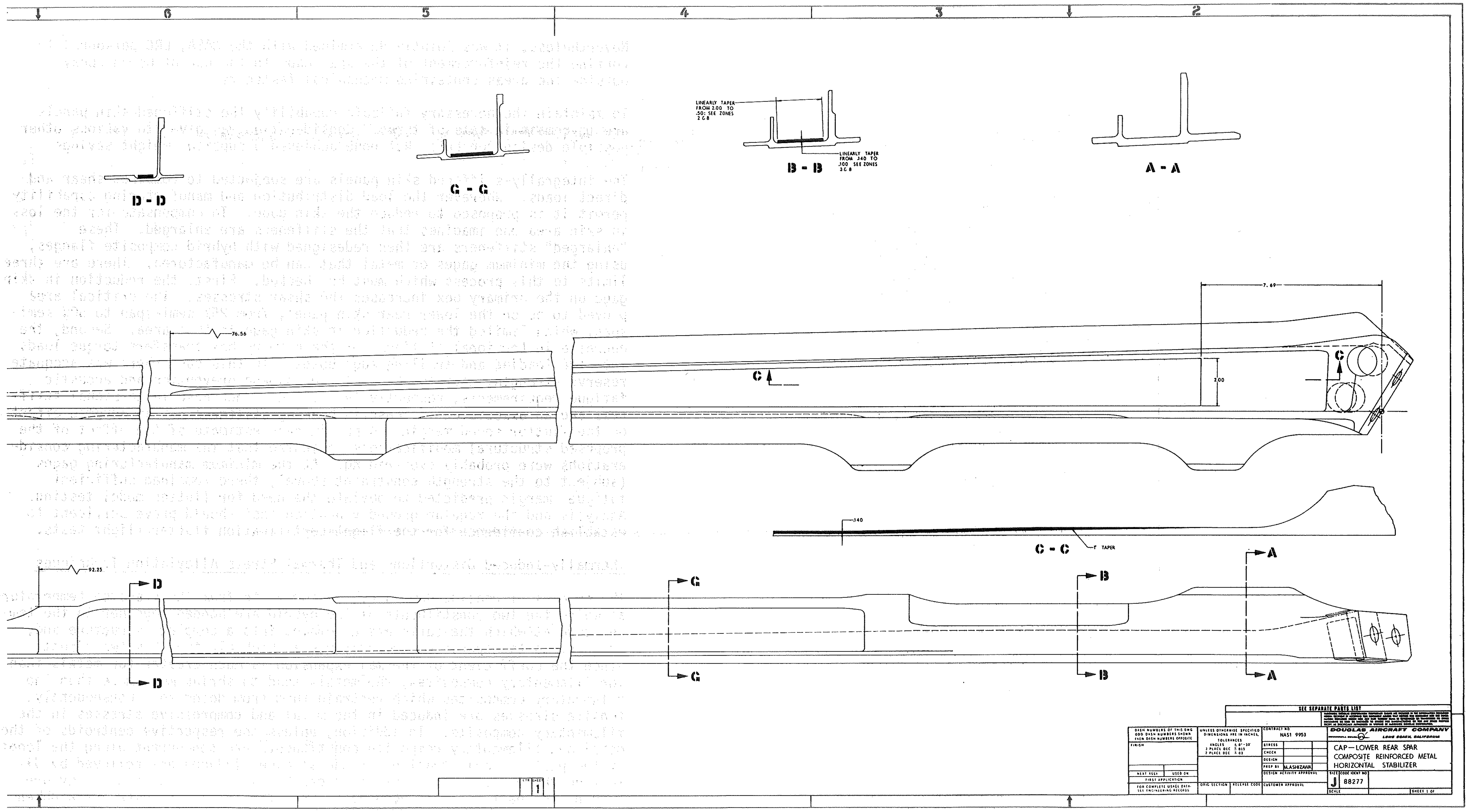


FIGURE 7. CAP-OUTBOARD SECTION LOWER REAR SPAR, COMPOSITE REINFORCED METAL (CONTINUED)

Nevertheless, it was jointly determined with the NASA, LRC personnel to confine the reinforcement of the spar caps to the use of boron-epoxy outside the areas containing mechanical fasteners.

To maintain the necessary failsafe capability the stiffened skin panels are to remain in sets of three. Consideration was given to various other possible design concepts, but none achieved a superior weight savings.

The integrally-stiffened skin panels are subjected to combined shear and direct loads. Wherever the load distribution and manufacturing capability permit it is proposed to reduce the skin gage. To compensate for the loss in skin area one imagines that the stiffeners are enlarged. These "enlarged" stiffeners are then redesigned with hybrid composite flanges, using the minimum gages of metal that can be manufactured. There are three limits to this process which must be checked. First, the reduction in skin gage on the primary box increases the shear stresses. The critical area proved to be on the lower rear skin panel, from 25% semi-span to 50% semi-span, which limited the reduction in skin gage in that area. Second, the decrease in torsional stiffness of the primary box transfers torque loads into the leading and trailing edge boxes. In this case there was adequate reserve strength, dictated by hailstone damage prevention and acoustic fatigue requirements, respectively. Third, a decrease in torsional stiffness and an associated increase in bending stiffness has an adverse effect on the flutter speed margin. A preliminary estimate of the effect of the proposed structural modifications indicated that the manufacturing considerations were probably over-riding. At the minimum manufacturing gages (subject to the strength constraint above), there remained sufficient fatigue margin predicted to obviate the need for flutter model testing. Analysis and the regular ground vibration test should prove sufficent to establish confidence for the flight-certification flutter flight tests.

<u>Thermally-Induced Distortions and Thermal Stress Alleviation Techniques</u>

The process of cooling down a hybrid composite from the elevated temperature at which the two constituents of the hybrid are bonded together to the temperature at which the parts are assembled into a complete structure and, further, to the operating temperatures at high altitude has two effects. Since the coefficient of thermal expansion is much greater for metals than for filamentary composites, the metals tend to shrink much more than the filamentary composites which restrain them from doing so. Consequently, tensile stresses are induced in the metal and compressive stresses in the filamentary composite. In addition, unless the respective centroids of the metal and filamentary composite constituents are concurrent along the length of the part, a small fraction of the stresses induced are relieved by distorting the part. * The distortion can be overcome in some cases by precurving the part prior to bonding so that the induced stresses straighten

*The distortion has proved to be predictable in both Boeing and Douglas tests. It should be noted, however, that the stress-free temperature is more closely related to the cure-initiation temperature of the adhesive than to the actual (possible higher) temperature at which the bond was cured.

it out, or by so designing the cross-sections that the respective centroids are nearly coincident (if necessary by adding extra material). However, the thermal stresses themselves can be prevented only by mechanically constraining the hybrid constituents to prevent their expansion at the bonding temperature. This is very difficult for tapered parts, or those with rapidly varying cross-sections, but certain techniques have been evolved for parts of (nearly) constant cross-section. The use of room temperature curing adhesives has been considered as a technique to eliminate residual stresses, but aerospace industry experience with such adhesives for metal-to-metal bonding has been unfavorable in terms of environmental degradation, fatigue behavior, and loss of strength with aging.

Bonded Joints

Those adhesive stresses arising from thermally-induced residual stresses are confined to a narrow strip around the periphery of the bonded region of the hybrid composite. Inside that boundary zone an essentially uniform stress state exists in the components of the hybrid. As the locked-in stresses in the adherends are reduced to zero at the free edges of the bonded region of the hybrid composites, the internal loads are reacted by shear stresses in the adhesive. The adhesive must also be capable of transferring part of the externally-applied load from the metal end fittings to the filamentary composite. Consequently the design of such joints is critical and calls for either a scarf or a stepped-lap joint. A uniform lap joint attempts to transfer the load over too short a distance, inducing premature failure. It appears that considerable analysis and development work needs to be done on joint design of full-scale structures, (scarf and stepped-lap) so that an efficient load transfer capability is achieved. Preliminary analyses indicate that compressive loading can be more severe than tensile loading and that there is a nonlinear scale effect.

2.1.4 Analysis

Previous analyses indicate that the all metal DC-8 horizontal stabilizer structure could suffer a complete failure of a single stiffened panel or single spar cap, in either the constant section or an outer section, and still carry Federal Aviation Administration failsafe loads. Failures were considered only in the upper elements because it is those that are subjected to fatigue cracking under the predominantly down-tail load. A similar failsafe analysis was also performed for the four-member attachment to the fuselage.

Loads

Horizontal tail loads had already been calculated for a large number of conditions including all parts of the maneuvering and gust envelopes to ensure that the maximum load for each part of the structure had been obtained. Such computations were conducted for all models of the DC-8 series. These analyses have confirmed that the stabilizers are completely interchangeable, between models with only one minor exception. With the introduction of the Super-60 series aircraft a more powerful operating screw jack was installed. This necessitated some local reinforcement of

Static Strength and Stiffness Considerations

This aspect of hybrid composite behavior has received considerable publicity because of the ease of analysis and the demonstrable benefits. The elastic stiffness of a uniaxial hybrid is governed by the law of mixtures:

$$E_H = E_M k + E_C (1-k) \tag{1}$$

where E is Young's modulus and k represents the volume fraction of metal. The subscripts H, M and C refer, respectively, to the hybrid, the metal and the filamentary composite. The density, likewise, is given by

$$\rho_H = \rho_M K + \rho_C (1 - K) \tag{2}$$

Because E_C is much higher than for most metals and ρ_C much less, the specific stiffnesses of hybrid are markedly superior to those of metal alone.

In computing the strengths of hybrids, it is necessary to take into account the thermally-induced locked-in residual stresses. Elementary thermal-stress theory predicts that for long specimens of uniform cross-section, the locked-in stresses are:

$$\sigma_{MR} = -\frac{[\alpha_M - \alpha_C]\,\Delta T}{\left[1 + \frac{(EA)_M}{(EA)_C}\right]} E_M, \qquad \sigma_{CR} = \frac{[\alpha_M - \alpha_C]\,\Delta T}{\left[1 + \frac{(EA)_C}{(EA)_M}\right]} E_C \tag{3}$$

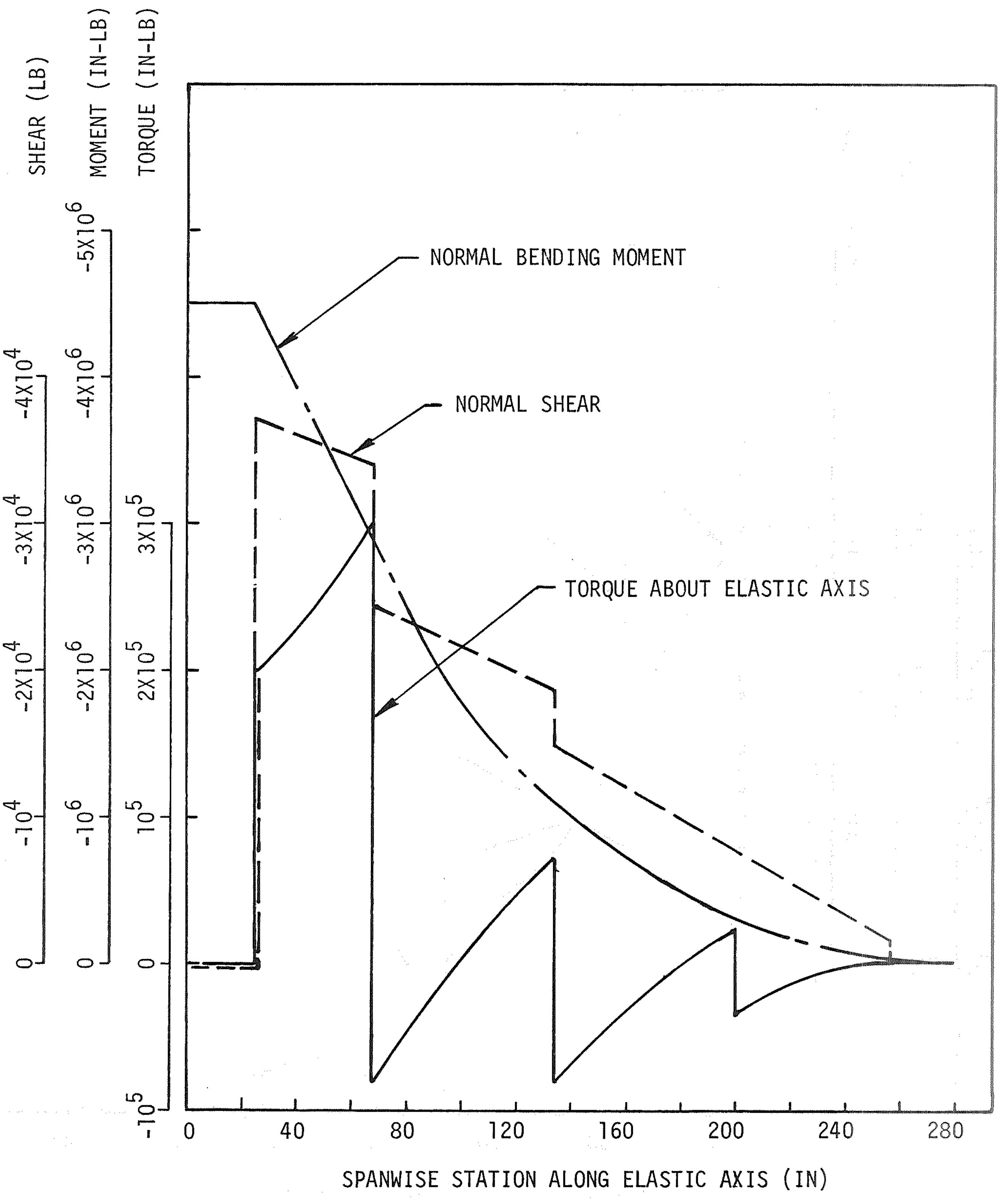


FIGURE 8. MAXIMUM BENDING CONDITION - DC-8 HORIZONTAL STABILIZER LIMIT LOAD DISTRIBUTIONS

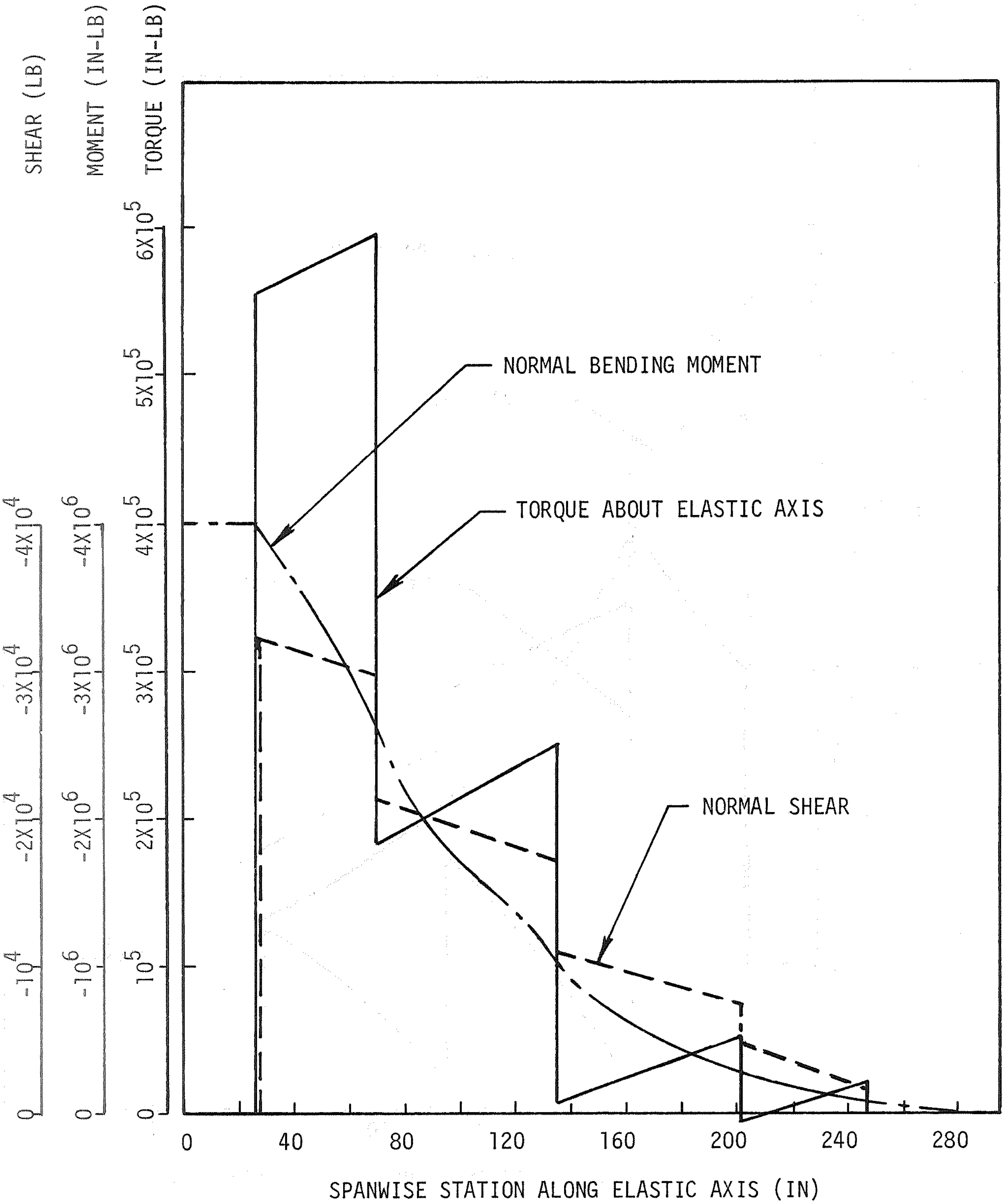


FIGURE 9. MAXIMUM TORQUE CONDITION - DC-8 HORIZONTAL STABILIZER LIMIT LOAD DISTRIBUTIONS

Generally, tensile stresses σ are induced in the metal (M) and compression in the filamentary composite (C) during cooling down from the curing temperature T_C to the operating temperature T_O. The temperature difference ΔT is defined as $T_O - T_C$. For most metals used in aerospace structures, the coefficient of thermal expansion α_M greatly exceeds α_C, as shown in Table I.

Figures 10 and 11 depict the theoretical static behavior of the aluminum/boron-epoxy hybrid bonded together at 250°F for a series of operating temperatures. It is evident that this is a "favorable" combination of materials inasmuch as the addition of any fraction of boron-epoxy to the aluminum represents an improvement in the specific properties of the total cross-section. Other combinations considered are not so favorable. Indeed, because of mismatch of the failure strains, whereby one constituent fails completely before the other is highly loaded, several combinations prove to be inferior to the metal alone on a specific weight basis. The "favorable" combinations can be deduced from the various stress-strain characteristics and include also titanium/low-modulus graphite-epoxy. Titanium/boron-epoxy appears at first to be an unfavorable combination in this regard, but the low mismatch of coefficients of thermal expansion overrides the strain mismatch.

In the actual static testing of hybrid composites under tension or compression it is not always possible to detect the locked-in strains. The reason for this is that the locked-in strains can relieve themselves internally once the metal has yielded under load. In the buckling of long columns also, (1) the residual stresses have no effect because the structure does not yield. The influence of thermally-induced locked-in stresses is observed most in fatigue loading and in joint strengths.

TABLE I

COEFFICIENTS OF THERMAL EXPANSION

MATERIAL	α, 10^{-6} IN./IN./°F
STEEL	6.0-6.3
TITANIUM	5.0-5.9
ALUMINUM	12.5-13.0
MAGNESIUM	14.0
BORON-EPOXY	2.7
S GLASS-EPOXY	3.5
GRAPHITE-EPOXY	0-1.0

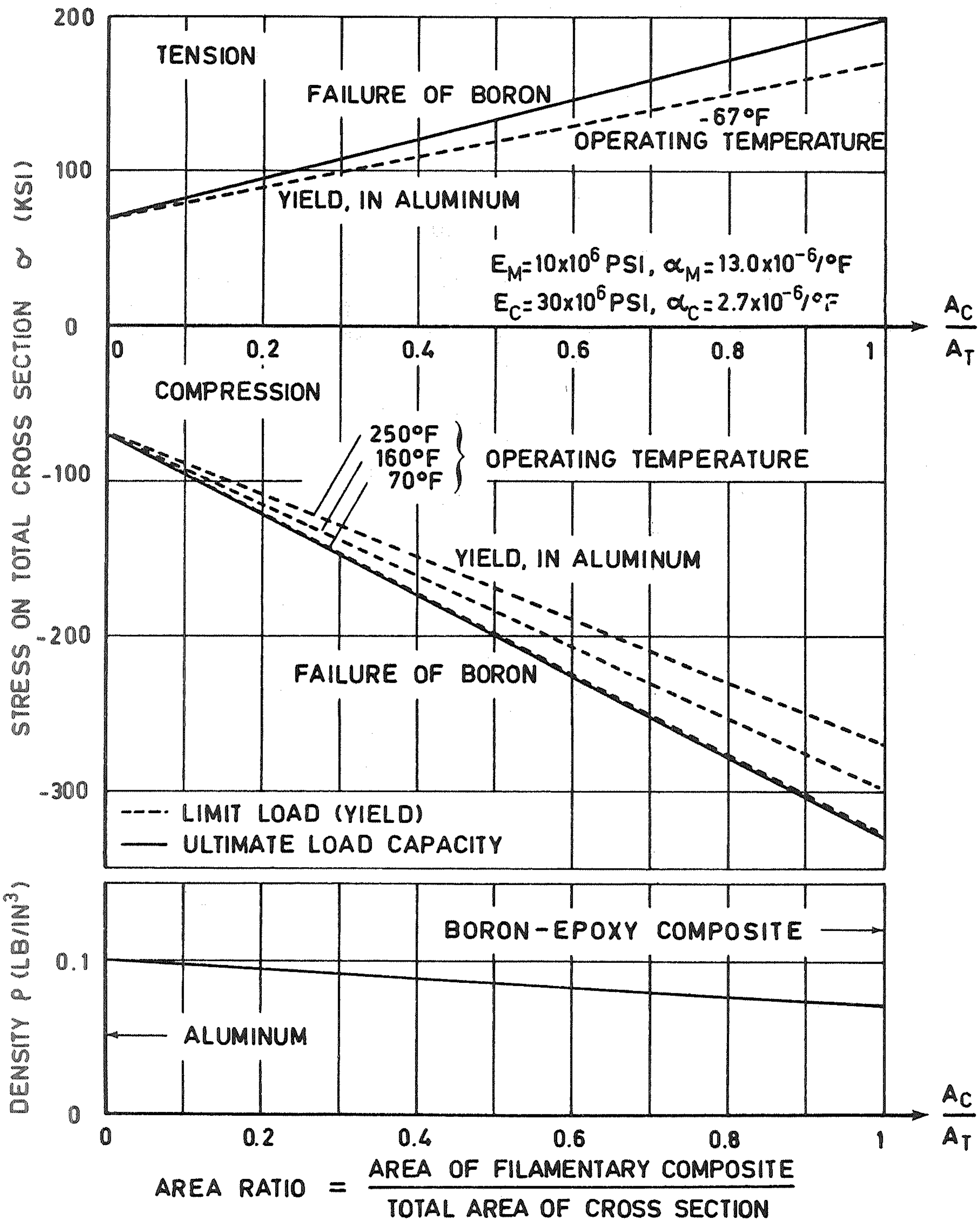


FIGURE 10. EFFECT OF THERMAL AND STRAIN MISMATCH ON STRENGTH OF HYBRID. 7075-T6 AL AND BORON-EPOXY COMPOSITE(50% FILAMENT VOLUME) CURED AND BONDED AT 250^{o} F.

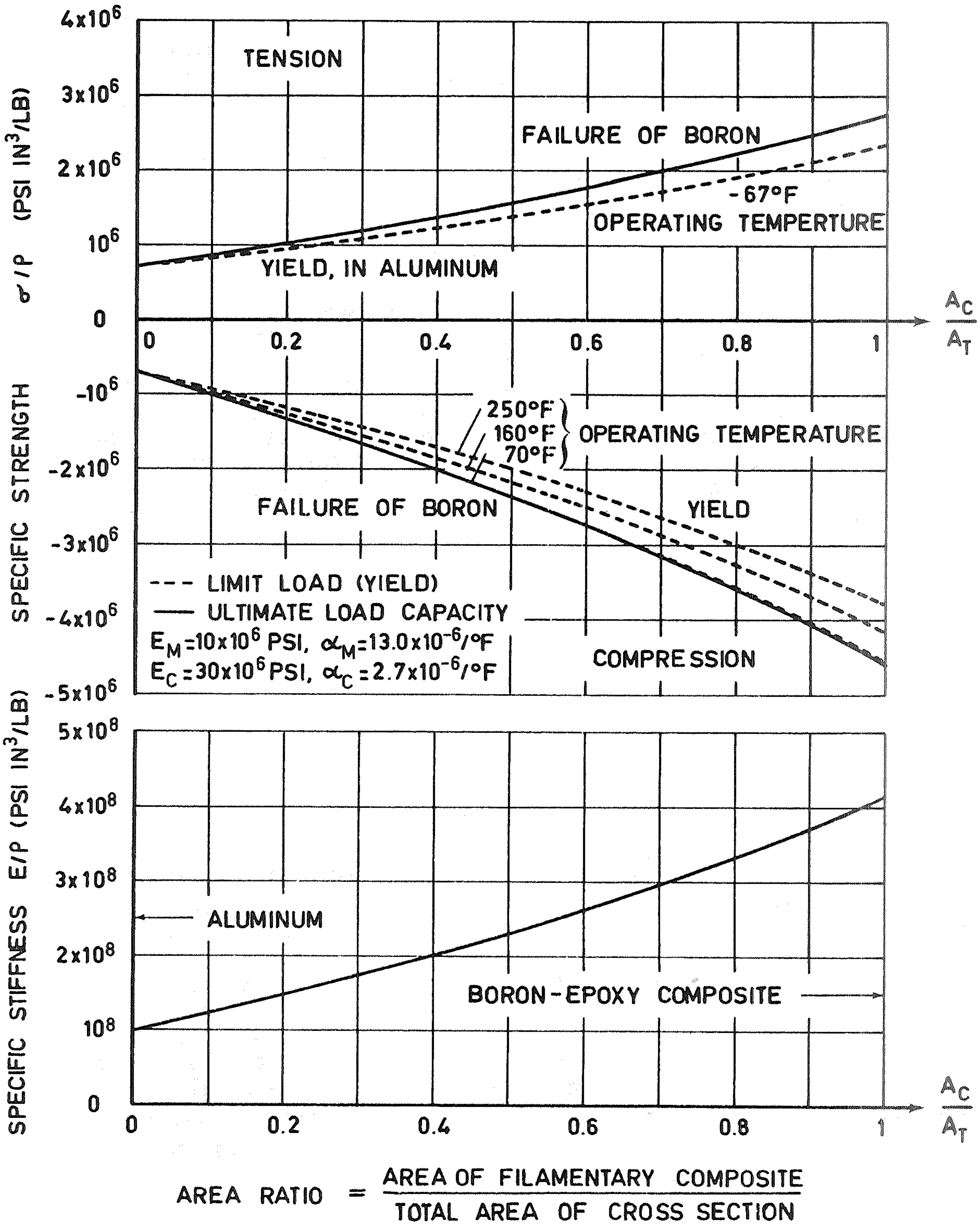


FIGURE 11. SPECIFIC PROPERTIES OF HYBRID COMPOSITE. 7075-T6 AL AND BORON - EPOXY COMPOSITE (50% FILAMENT VOLUME) CURED AND BONDED AT 250° F.

Fatigue Behavior

The fatigue behavior of filamentary composites is vastly superior to that of metals. The residual stresses in the hybrid aggravate this imbalance by inducing tension in the metal and compression in the composite. Consequently, the metal is prone to fatigue failure, not the filaments. An idea of the magnitude of such residual stresses can be gaged from Figure12 for the the aluminum/boron-epoxy system: typical thermally-induced locked-in stresses range from 10 to 20 ksi. The effect of such stresses on the fatigue life of hybrid composites is significant and requires careful consideration in the design and analysis process.

Figure 13 illustrates the mechanism whereby the tensile residual stresses inevitably either decrease the load capacity of the metal for a constant life or decrease the life for maintaining the same load. All characteristics illustrated for the varying locked-in stresses are for an external cyclic load for which R = + 0.2. Preliminary analyses indicate that Goodman (constant life) diagrams are suitable for the prediction of fatigue lives of hybrid composites. Figure 14 depicts the weight saving potential of uniaxial hybrid composites with fatigue critical structures. The percentage weight savings illustrated are for structures carrying the same loads for the same number of cycles to failure. They derive from idealized Goodman diagrams via the formula:

$$\text{Relative Weight} = \frac{E_M \times \rho_H}{E_H \times \rho_M} \times \frac{\sigma_{ULT_M}}{\sigma_{ULT_M} - \sigma_{RESIDUAL_M}} \qquad (4)$$

where the ultimate metal stress, on the gross cross-section of a structure including mechanical fasteners, is typically 60 ksi for 7075-T6 aluminum. This formula holds for essentially all numbers of cycles to failure and all R values of loading for both hybrids except that it is slightly conservative for aluminum under low-cycle fatigue in combination with reversed loading. The weight savings predicted by equation (4) are optimistic because they omit an allowance for the weight of the additional joints for transferring the loads into and out of the filamentary composite reinforcement. Figure 14 indicates the order of magnitude of the weight savings possible for fatigue-critical structures. Generally what limits the amount of reinforcement used is the shear load present in addition to the axial load. A typical temperature differential for subsonic aircraft is 300°F. A weight savings of 10% using the aluminum/boron-epoxy hybrid would therefore require at least 14% of the cross-section to be boron-epoxy.

In the case of the proposed hybrid composite horizontal stabilizer an analysis was performed of the cumulative fatigue damage due to gusts and ground-

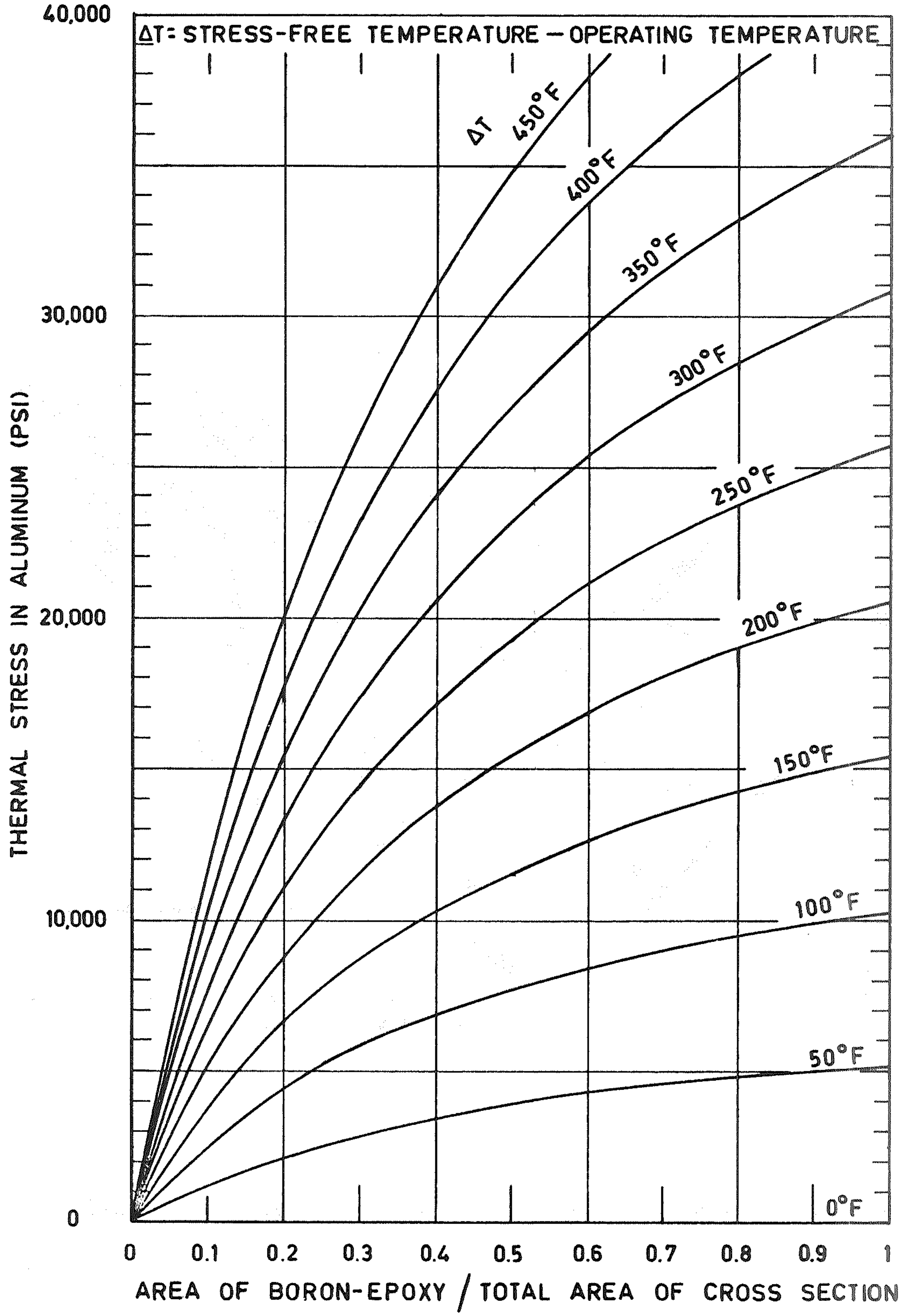


FIGURE 12. THERMAL STRESSES IN ALUMINUM/BORON-EPOXY HYBRIDS

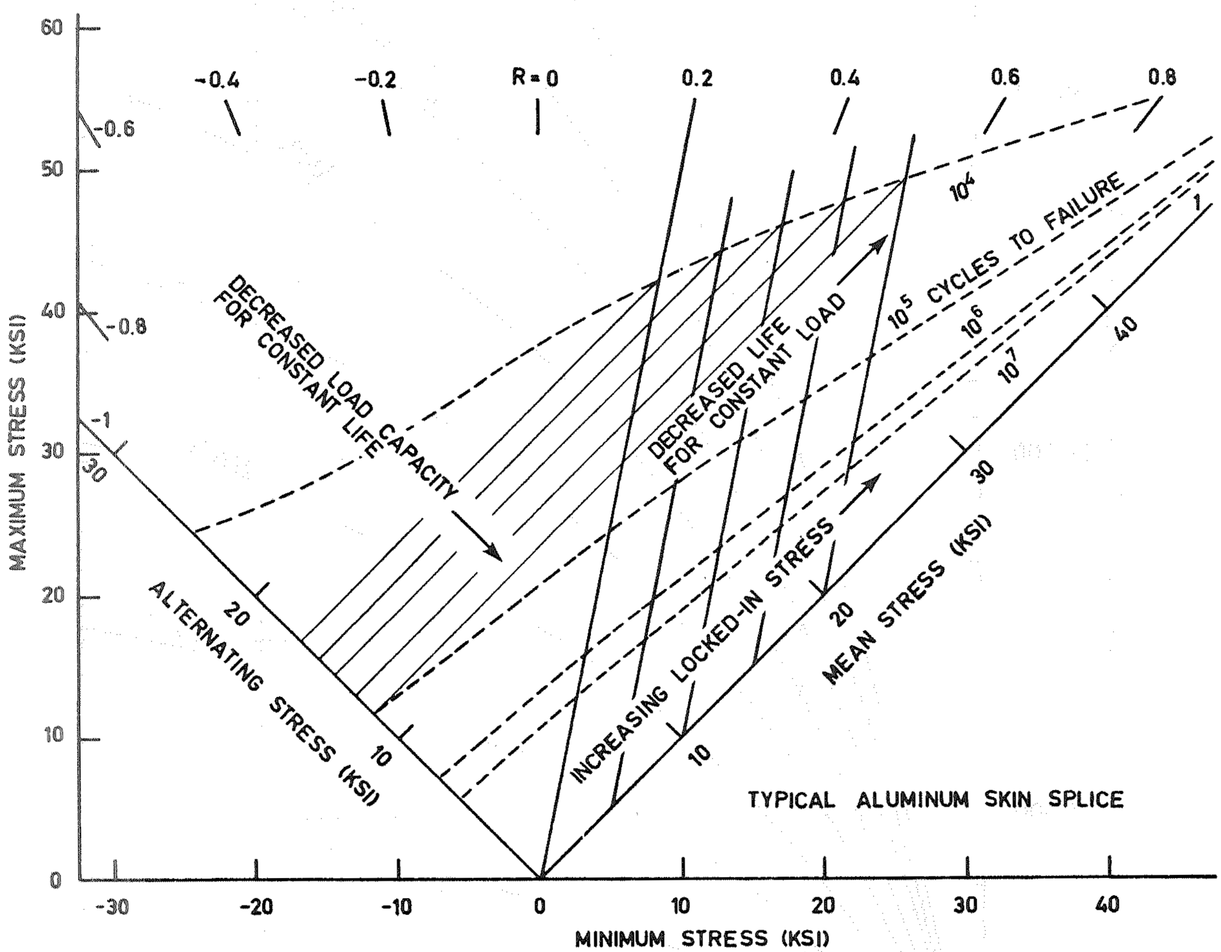


FIGURE 13. REDUCTION IN FATIGUE PERFORMANCE DUE TO LOCKED-IN STRESSES

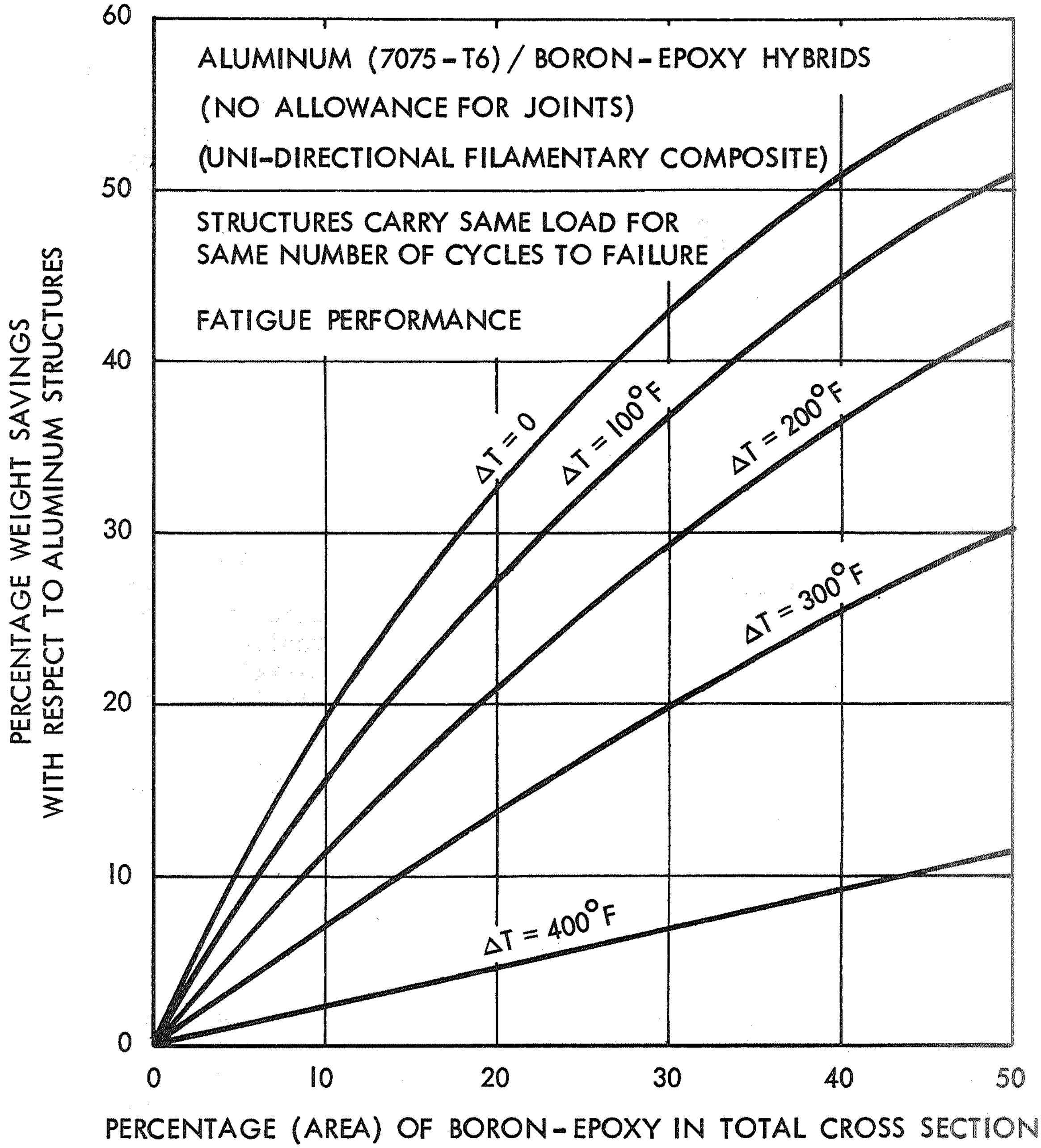


FIGURE 14. WEIGHT SAVINGS POSSIBLE IN FATIGUE-CRITICAL ALUMINUM/BORON-EPOXY HYBRID STRUCTURES.

air-ground cycles. This revealed that the damage would not be sufficient to render the design critical in fatigue with 250°F curing adhesives. The use of 350°F curing adhesives would impose the need for additional material to reduce the fatigue stresses. Based on the existing DC-8 analysis, it can be stated that 70% of the damage accrues mainly as the result of gust loading, with ground-air-ground cycles imposing practically all of the remaining 30%. It is necessary to allow for the different temperatures at which the maximum and minimum stresses are sustained because they occur at different periods in the flight and are associated with different locked-in stresses. To assume an average operating temperature underestimates unduly both the maximum and minimum stresses. Horizontal stabilizers are normally not sensitive to fatigue damage because the maximum design loads are so much greater than the regularly occurring flight loads.

Preliminary Flutter Analysis

A preliminary flutter analysis was performed on the basis of the estimated reduction in torsional stiffness, shown in Figure 15. Only the fundamental bending-torsion case was examined at this time. The object of this investigation was to assess the likely feasibility of the proposed changes by comparison with the known behavior of the all-metal horizontal stabilizer. Two configurations were analyzed. In the first, a design was proposed which is believed to be realistic while, in the second, the shear material had been removed entirely to the extent permitted by static strength and manufacturing requirements. Both configurations were predicted to provide a positive flutter margin. The reduction of shear material in the constant section influences the flutter behavior in a single failure condition with one of the four support points failed. The associated mode is asymmetric and involves much greater analysis effort. It is among these analyses proposed for the detail design and analysis phase of this program which are necessary for flight certification.

Adhesive Stresses in Hybrid Scarf Joints

The analysis of the scarfed joints at the end of the filamentary composite reinforcement is more complex than for a scarf joint between identical adherends. In the latter case, a uniform shear stress is developed along the entire length of the joint with minor strain variations confined to each end. The hybrid composite scarfed joint, on the other hand, may include stiffness imbalance, as well as thermal mismatch. Both these factors cause a significant and continuous variation in the adhesive shear strain along the length of the joint. The two factors may nullify each other or be compounded but, if they alleviate the adhesive shear strain concentration at one end of the joint, they must inevitably aggravate the concentrations at the other. Also, the joint strength will depend on whether the load be tensile or compressive. The analysis of such joints rests on the complete stress-strain characteristic of the adhesive in shear, with the ultimate shear strain providing the failure criterion. Figure 16 illustrates these phenomena for a stiffness-balanced, but thermally mismatched, hybrid scarf joint. As the load is progressively increased the fully-stressed portion of the joint increases while the maximum strain in the adhesive at the

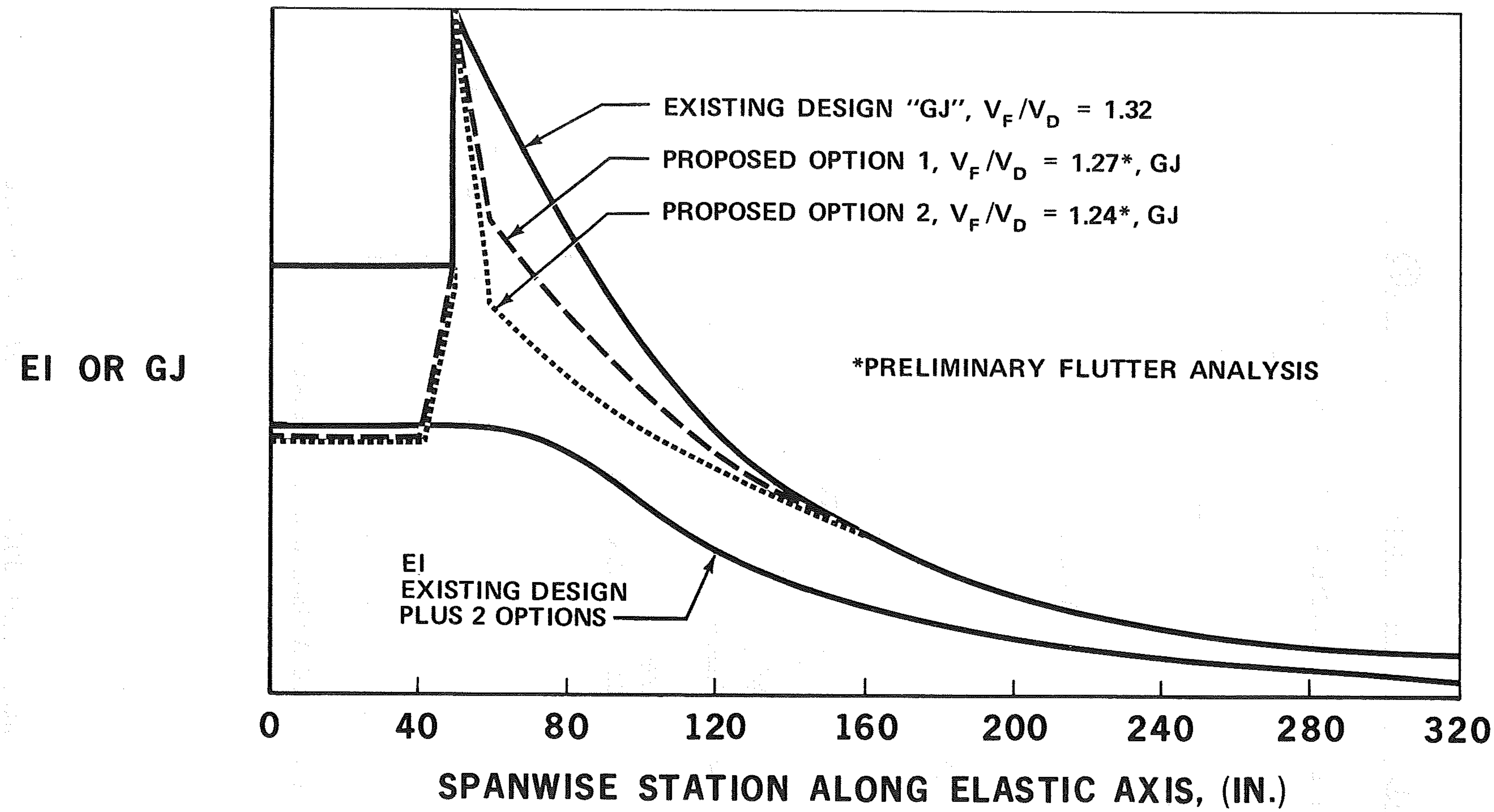


FIGURE 15. DC-8 HORIZONTAL STABILIZER RIGIDITY

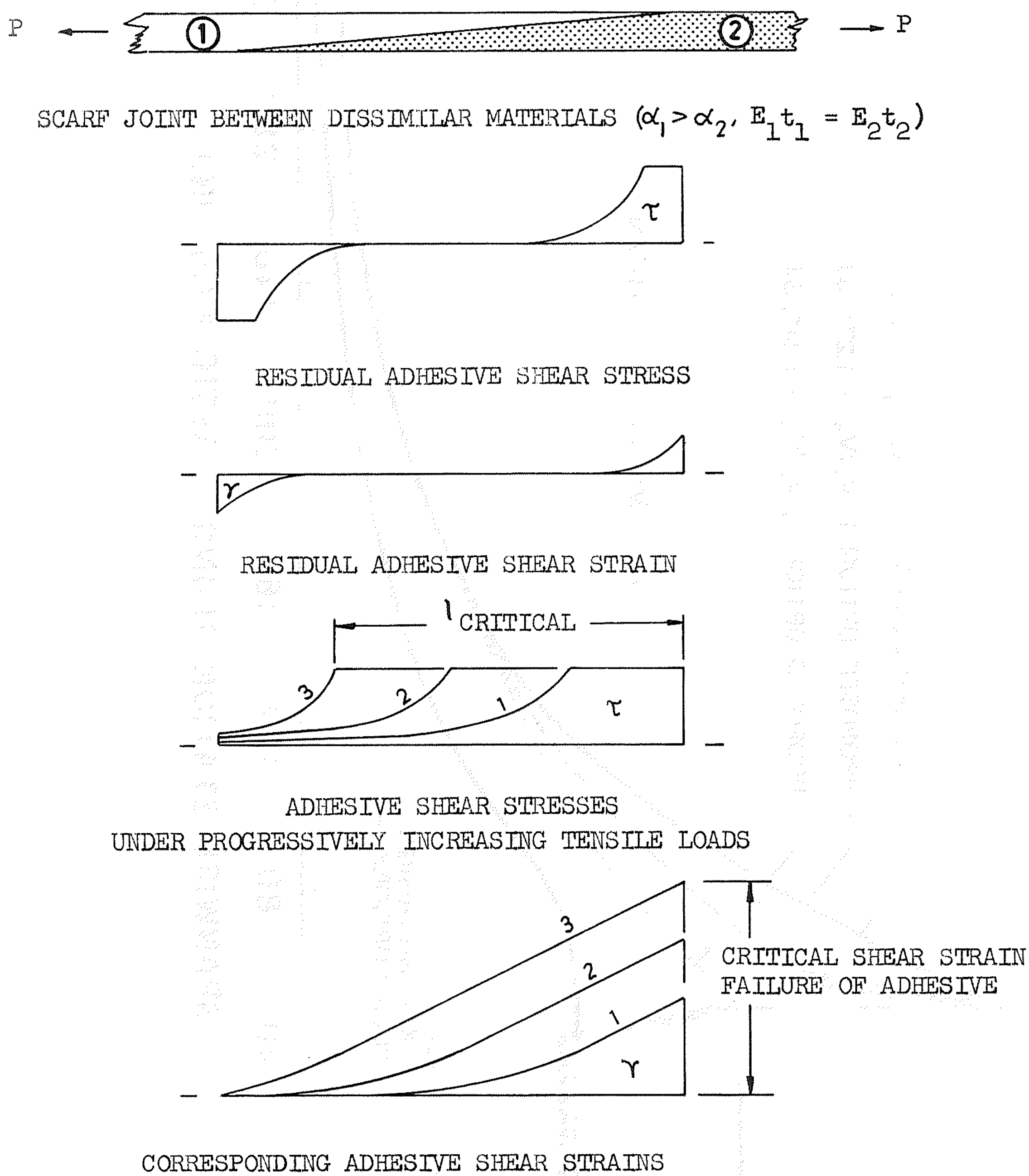


FIGURE 16. BONDED SCARF JOINT WITH THERMAL IMBALANCE

critical end also increases. If the filamentary composite is sufficently thick, the joint will fail as shown before the composite is fully loaded. The analysis of such joints indicates that a markedly nonlinear geometric scale factor prevails and that subscale tests are not representative of the behavior of a full-scale structure. Further experimental studies of these problems appear necessary in the light of the preliminary analytical predictions.

2.2 FABRICATION METHODS

A study of the tooling, fabrication, and assembly requirements for the manufacturing support of the hybrid composites horizontal stabilizer program indicated several major areas of importance. Special attention will have to be directed to the problem of potential warpage of hybrid composite elements resulting from the elevated temperature bonding of dissimilar materials with significantly different thermal coefficients of expansion. See Section 3.2, Design and Analysis for a detailed discussion of this topic.

The integrally stiffened skin panels for the existing all metal design are 0.045 inch (minimum) thickness over approximately 30% of the panel area for the center and rear panels. The total area of the panels to be machined to minimum thickness for the hybrid composites configuration is about 40%. The machining of the aluminum skins to 0.045 inch over a much larger percentage of the area may cause difficulties in maintaining the desired thickness tolerances and in subsequent handling and joining operations resulting from part warpage and surface flexibility.

SECTION 3

DEMONSTRATION PROGRAM PLANS

This section of the report describes two Program Plans to demonstrate the feasibility of hybrid composite structural members using a DC-8 horizontal stabilizer.

The recommended plan, which consists of a total of five phases including this Phase I Study, involves three horizontal stabilizers (two flight service and one ground test). Technical personnel of the NASA, LRC suggested at the conclusion of the Phase I Study that a five phase plan including three horizontal stabilizers be considered. The Douglas Company presented the five phase plan as the preferred choice at the Phase I Final Review. The alternate plan, which consists of a total of three phases including the Phase I Study, involves one horizontal stabilizer for test and subsequent flight service. The alternate plan was the basic plan throughout the Phase I Study and therefore is reported in a detailed manner.

The two plans are similar in that both progress from design allowables and certification (FAA) specimen fabrication and testing to detail design and analysis of the hybrid composite stabilizer together with subcomponent testing followed by the fabrication and assembly of the stabilizer(s). Subsequent program milestones include static proof test(s), a fatigue test (Recommended Plan only), installation of the stabilizer(s) on the DC-8 airplane(s), a ground vibration test (GVT) of the stabilizer and a flight certification period to satisfy FAA requirements for flutter margins and aerodynamic stability and control characteristics. After FAA certification, the airplane(s) will be returned to the airline for flight service for the Recommended and Alternate Plans, two years and one year, respectively. Finally, the airplane(s) will be refurbished with the original all metal horizontal stabilizer(s).

The Demonstration Programs, which include the Phase I Study, are outlined as to the additional phases in each of the plans.

The Five Phase Program

Phase II - Advanced Development Tests

A review will be made of the current state-of-the-art, several reinforcing structural concepts will be evaluated and the best concept selected. A test program will be conducted for (1) materials selection, (2) design allowables, and (3) substantiation and verification of the bonded joint representing the inboard end of the outboard structural assembly (Figure E).

Phase III - Detail Design and Analysis

Detail design and analysis of the skin panels and spar caps for strength, stiffness, shear buckling, and column stability will be accomplished. A trade study and analysis will be conducted on scarfed and stepped joints. An analysis will be made of locked-in thermal stresses and their effect on the strength and fatigue life of the hybrid composite structural members. A subcomponent test program will be conducted to verify and substantiate the design of the panels and spar caps (Figure F).

Phase IV - Component Fabrication and Test Plan

Tool design and fabrication and manufacturing processes will be completed. Three horizontal stabilizers will be assembled with one of the three stabilizers instrumented with flight flutter instrumentation. A test plan for static proof testing of three stabilizers and the fatigue and ultimate strength testing of one of the stabilizers will be completed and test drawings furnished (Figure G).

Phase V - Component Tests and Flight Service

Each of the three stabilizers will be static tested to proof load (100% limit load). In addition, the ground test article will be tested for fatigue life and ultimate strength. Two of the composite reinforced horizontal stabilizers will be shipped to UAL's maintenance base in San Francisco and installed on DC-8 airplanes. The all metal horizontal stabilizers will be held in storage at the airlines maintenance facility for the duration of the flight service period. One of the installed stabilizers will be given a ground vibration test. Flight acceptance checks will be required for both airplanes with one of the airplanes participating in a flight certification period. After FAA certification, the airplanes will be returned to the domestic airline for a flight service period of two years minimum. Finally, the airplanes will be refurbished with the original horizontal stabilizers (Figure H).

The Three Phase Plan

Phase II - Design, Analysis, Test, and Component Fabrication Program

Phase II will consist primarily of detail design and analyses of the horizontal stabilizer supported by design allowables and certification (FAA) specimen fabrication and testing. In addition, the full scale hybrid composite horizontal stabilizer will be built and statically tested.

Phase III - Component Test and Flight Program

Phase III will consist of several tasks such as installing the hybrid composite stabilizer on a DC-8 airplane followed by a GVT of the stabilizer and a flight certification period. After FAA certification, the airplane will be returned to the airline for use in regularly scheduled airline operations for a period of one year. Finally, the airplane will be refurbished with the original all metal horizontal stabilizer.

3.1 MATERIALS

3.1.1 Materials Approach

During the initial phart of Phase II, a survey of the latest literature will be made while contacts are maintained with material suppliers. Although this effort will probably be small, the continued improvements in available products make it desirable to obtain the most recent best materials for evaluation and incorporation into the hardware phase of the program. The final selection of materials will be made after the initiation of Phase II so that the advantages of the latest breakthrough in either cost or materials technology can be maximized for this program. The detail definition of the first group of specimens to be fabricated and tested will be determined as soon as possible after initiation of Phase II. This will be followed by estimates of materials requirements for both boron-epoxy and the adhesives so that materials delivery can be expedited. Materials analysis including costs are discussed in Appendix C. A new concept called bi-adhesive bonding is discussed in Appendix D.

All aspects of materials usage, handling, and storage will be monitored to insure that proper care be given to the filamentary prepregs, adhesives, and metal parts for the timely fabrication and test of all necessary allowables data and certification specimens. Specifications will be issued, documentation will be maintained for fabrication steps, and quality control procedures implemented. These important aspects are discussed in detail in Appendix E.

3.1.2 Processing Considerations

The composite materials (prepreg and adhesive) must be capable of reliable, repeatable processing while maintaining high mechanical properties.

Two different methods of composite manufacture are proposed. First, composite laminates will be precured, and secondarily bonded in place. Second, the composite will be laminated and bonded in place in the one cure cycle.

Careful consideration must be given to cleaning techniques and preparation of the aluminum surface for bond for both fabrication procedures. The secondary bond fabrication method will require a careful cleaning operation of the precured composite prior to the bonding operation. The adhesive system for secondary bond must cure at as low a temperature as possible and still obtain satisfactory mechanical and physical properties.

The co-cure and bond fabrication (second system) must utilize an adhesive and a fiber matrix combination that are compatible with each other in regard to curing cycle and chemical reaction. These materials must cure at as low a temperature as possible to maintain satisfactory mechanical and physical properties to minimize the dissimilar thermal coefficients of expansion that exist between Boron-epoxy and aluminum. See Table I. Curing and bonding in place will be done wherever possible to obtain significant savings in labor and machining costs compared to secondary bond fabrication.

3.2 DESIGN AND ANALYSIS

3.2.1 Design Approach

Several of the most promising hybrid composite structural concepts will be evaluated and the most promising integrally stiffened skin panel and spar cap configurations selected for detailed design. Design optimization will be accomplished on the structural elements and the overall structure with proper consideration for the critical loading conditions and failure modes. Joint design studies will be conducted for the most efficient design to be incorporated into the hybrid composite elements for transferring loads into and out of the composites, e.g., stepped lap and scarf joints will be evaluated.

A complete test plan will be coordinated with all cognizant departments and necessary drawings and documents provided. Test drawings will be issued for the fabrication, assembly, and test of the basic data allowables and subcomponent specimens.

Weight changes to structural elements will be established and weight and balance changes to the DC-8 airplane determined.

Engineering drawings will be completed for all production details, assemblies, and installations and released for tooling, scheduling, and manufacturing. Finally, the end-item test drawing(s) for the static proof test (100% limit load), fatigue test, and ultimate strength test will be provided. The last two tests are applicable only for the recommended Demonstration Program.

3.2.2 Analysis

Approach

Initially the design will be reviewed in the light of improvements in the state-of-the-art prior to the commencement of Phase II. The analysis method and procedures will be updated and improved, as necessary, to handle this hybrid composite design. The design criteria employed will remain the same as those employed for the metal horizontal stabilizer. The critical design conditions are: (1) Maximum down tail load, (2) maximum up tail load, and (3) maximum down tail load with the elevator at a large incidence. In addition, a variety of failsafe conditions must be examined for FAA certification. The basic horizontal stabilizer stress analysis is for static conditions. Because of the proposed reduction in torsional stiffness, some internal redistribution of loads will occur. The outer sections of the horizontal stabilizer are three-cell torque boxes, of which the front and rear boxes are not to be altered. The effect of the proposed modifications on the overall stiffness will therefore not be great. In assessing the effects on the stresses, however, a complete reanalysis will be necessary because of the torque loads transferred from one cell to another and the redistribution of stresses which will occur.

The various modified concepts proposed will be analyzed and modified as necessary, in the light of the results of the design data and subcomponent testing. The strength test requirements for the various component and subcomponent tests will be formalized early in Phase II of the program. Because the boron-epoxy reinforcement is to be applied longitudinally only, the analysis will not be rendered much more complicated than is the practice for all-metal designs, with two exceptions. The residual thermal stresses influence significantly the fatigue behavior and the composite-to-metal bonded joints will require more refined analysis to ensure a satisfactory design. Analyses will be performed on the modified (hybrid) structure to ensure that the design is satisfactory for flutter. (Preliminary analysis for the symmetric load case has indicated a small positive flutter margin in excess of 1.2 V_D for the proposed modifications). In addition, flutter computations are to be performed for a variety of failsafe conditions, which may impose a limit on the reduction in skin gage possible for the constant section.

The maintenance of aerodynamic control effectiveness will be monitored during the design by estimating the pertinent horizontal stabilizer aeroelastic parameters. In turn, the effects of changed aeroelastic characteristics on the critical stability and control characteristics of the airplane will be estimated. These critical conditions are: (a) static longitudinal stability in climb configuration, (b) static longitudinal stability in cruise, and (c) controllability with mistrim between V_{Mo}/M_{Mo} and V_D/M_D. On the basis of the flight tests the estimated stability, control and flutter characteristics of the airplane will be confirmed or revised, as necessary, in the major affected areas.

3.3 MANUFACTURING

3.3.1 Manufacturing and Tooling Approach

Design of new tools which are required for the fabrication and assembly of test specimens and the horizontal stabilizer will be initiated on the basis of preliminary information available from the design task. An assessment will be made of the general area to be established for the sheet metal forming, machining, assembly and bonding operations. Long lead item requirements will be established and affected groups notified as to the criticality of the different detail items and assemblies.

Tooling for this program will be based on the "quick-fix" or one use technique. The majority of the sheet metal tools for forming the ribs, stiffeners, and clips will be of hard wood or Masonite construction with the parts hand formed or hydroformed in the "O" condition for subsequent heat treatment. Drilling and trimming of sheet parts will be accomplished from layout rather than costly drill jigs and machining fixtures. Machined metal plates, forgings and extrusions will require more extensive tooling.

Assembly tools will of necessity be precision designed fixtures. The stabilizer box section will require an Assembly Jig; several subassembly jigs will also be required for proper alignment of spars, ribs, stiffeners, and clips and caps during their assembly.

3.3.2 Fabrication of Hybrid Composites

Fabrication Procedures

Two basic fabrication procedures will be used to fabricate all data and subcomponent specimens. These same procedures will be used for the fabrication of the DC-8 horizontal stabilizer demonstration structure. The two procedures will consist of (1) pre-curing of the boron reinforcement with a secondary bond of composite to metal, and (2) the layup, cure, and bond of the composite to the metal in one primary cure cycle.

Procedure A: Secondary Bond of Boron to Aluminum

The required number of plies of boron/epoxy prepreg will be laminated on a suitable surface coated with a release agent. Edge dams will be placed around the circumference of the laminate. Permeable separator film will be placed over the top of the laminate followed by the required amount of bleeder material, (one ply of bleeder for every four plies of boron prepreg). A non-permeable film, e.g. teflon or mylar will then be placed over the laminate and edge dams and sealed using masking tape. This will control the resin flow exclusively into the bleeder material and control final panel resin content. A pressure plate, the size of the laminate will be placed in position and the completed assembly will be sealed in a vacuum bag. See Figure 17 for details.

The layup will be cured in the autoclave using the required prepreg cure cycle.

BORON MATERIAL	AUTOCLAVE CURE CYCLE
3M Co, SP-295	2 hours at 250°F/85 psi 2 hours at 250°F-Post cure
AVCO RIGIDITE 5505	2 hours at 200°F/100 psi 2 hours at 300°F/100 psi 2 hours at 350°F/100 psi

Quality control coupons will be taken from the cured laminate and tested to determine laminate acceptability. Upon acceptance of the cured panel it will be prepared for bonding by an acceptable surface preparation method. The aluminum will be prepared for bonding by a series of operations namely; alkaline cleaner wash, water rinse, sodium dichromate sulfuric acid etch, final water rinse, and finally forced air dried in an oven.

The bonded area on the metal and cured panel will be coated with a corrosion inhibitive adhesive primer. After drying, a 250°F cure adhesive film will be laid in place and the boron panel positioned over it. The whole assembly along with prepared metal to metal adhesive control coupons will be bagged and cured in an autoclave for a minimum of one hour at 250°F and 100 psi. Upon completion of the cure, the quality control coupons will be tested. The assembly will be machined to final dimensions as soon as test results verify that the adhesive bond meets or exceeds the minimum requirements for strength.

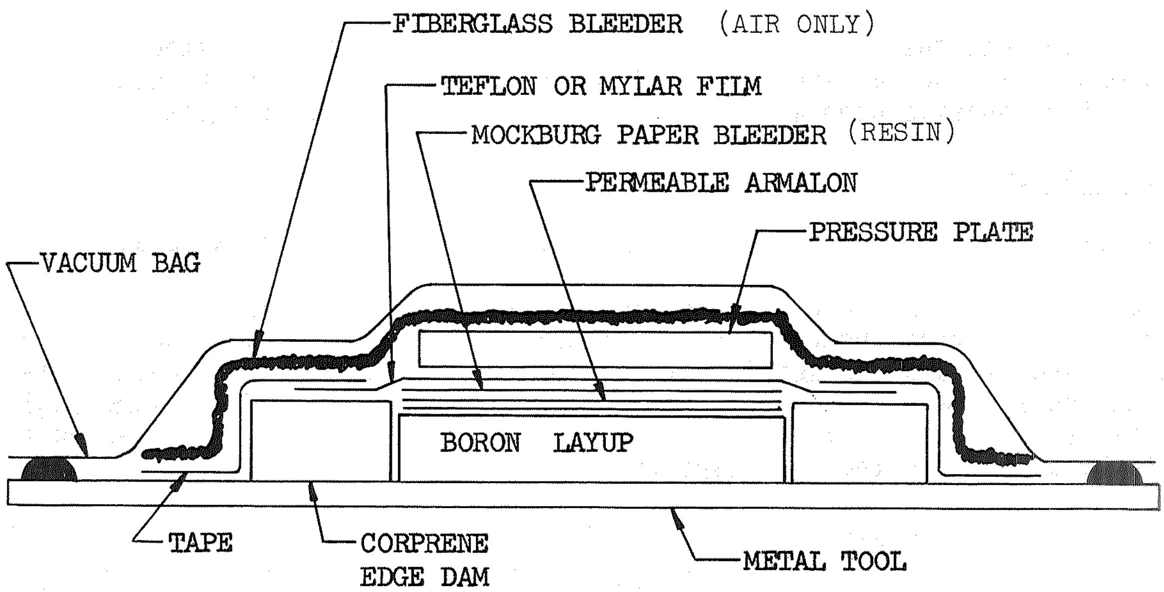


FIGURE 17. BLEEDER SYSTEM FOR BORON PARTS

Procedure B: Primary Cure and Bond of Boron to Aluminum

In this procedure, the bonding of metal to laminate and the curing of the laminate are accomplished in one process. The machined aluminum details are prepared for bonding exactly as described under Procedure A. The corrosion inhibitive adhesive primer will be applied to the bond area of the metal. A 250°F adhesive film will be placed on the prepared metal bond surface. 3M Company SP-295 boron prepreg will be laminated with the required number of plies on the area to be bonded. A quality control panel and metal to metal adhesive control coupons will be prepared for all panels cured. Edge dams will be located, separator film positioned, bleeder material laid, a non-permeable film sealed in place, pressure plate added, and the entire assembly sealed in a vacuum bag. This sequence of steps is identical to Procedure A. The layup will be cured in the autoclave using the following cure cycle: two hours at 250°F and 100 psi and two hours at 250°F post-cure.

After curing, the quality control coupons will be cut and tested. The assembly will be machined to final dimensions as soon as test results verify that the adhesive bond meets or exceeds the minimum requirements for strength.

Fabrication of Horizontal Stabilizer Details

All of the boron-epoxy composite material will be prefabricated as follows:

The Rear Spar caps have unidirectional strips of boron-epoxy composite bonded to the web surface between the stiffeners. The transition between the ends of the composite and the metal spar bathtubs at the inboard end, being a taper, requires a female layup tool using a transition section of the same angle. This tool is steel and will be adaptable to the various lengths required on the skin planks as well as the spar caps (See Figure 18).

The Outboard skin panels have unidirectional boron-epoxy strips bonded at the root (fillet) of the Tee sections. The details will be made in a special tool to obtain the tapered end design (See Figure 19). Special tooling will be required to obtain the tapered end design in precuring the web cap reinforcement boron laminates. These laminates will be secondarily bonded in place, with the laminate contour matched to the part contour, following standard bonding procedure. Primary curing and bonding will be done following the previously mentioned standard procedure.

Bonding is accomplished in the autoclave under pressure and temperature onto aluminum surfaces prepared by chemical cleaning. The composite materials are grit blasted to remove molding gloss and any mold release finishes.

Inspection of each phase of fabrication will insure adherance to blueprint design and dimensional tolerance. Quality Assurance will maintain surveillance during the subassembly techniques in riveting, bolt torquing and bonding by use of finished assemblies.

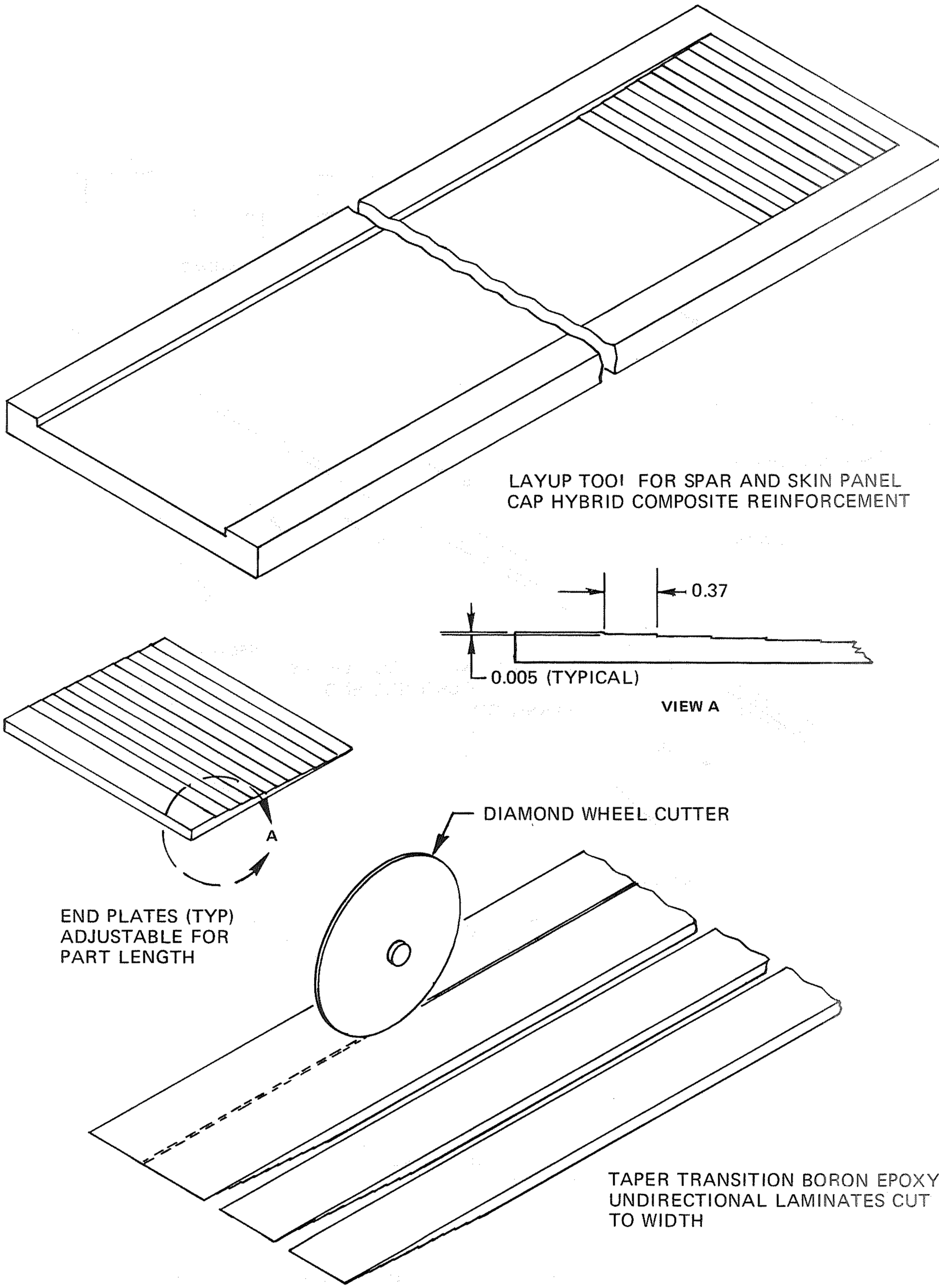


FIGURE 18. LAYUP TOOL FOR SPAR CAP AND PANEL LAMINATES WITH TAPERED ENDS

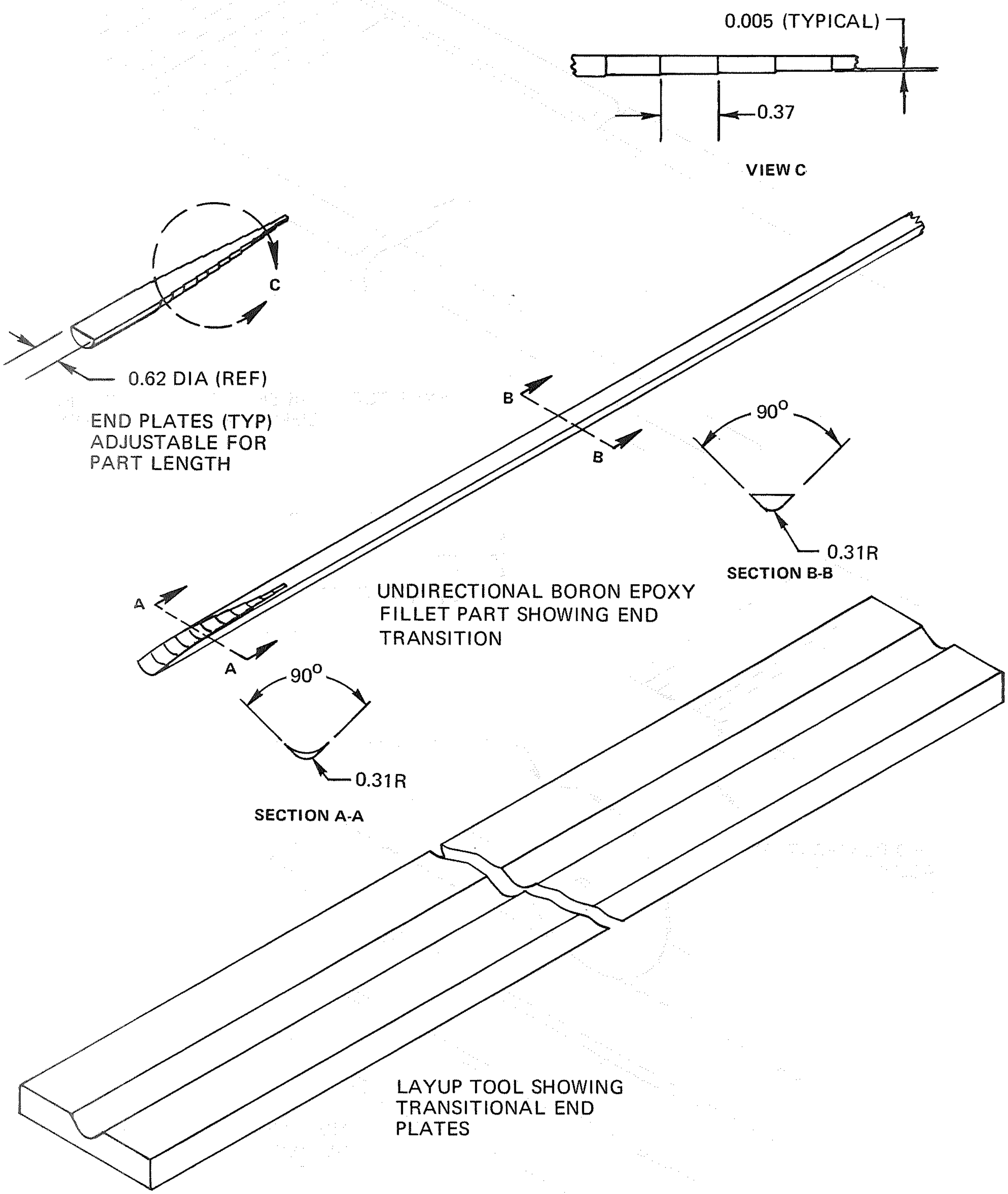


FIGURE 19. LAYUP TOOL AND CURED UNIDIRECTIONAL FILLET DETAIL

3.3.3 Production Processes and Assembly

The left and right trailing edge assemblies will be removed from an existing used stabilizer and reinstalled on the hybrid composite box section. The trailing edge subassembly is mechanically fastened with rivets and bolts to the box section. After the rivets in the skin panels are removed, the trailing edge will be clamped to a holding and assembly strongback. The bolts will then be removed and the trailing edge temporarily set aside for later use. The trailing edges are bolted and riveted into place as subassemblies and with careful removal should be usable without alteration. The standard removal and installation procedures which are outlined in the appropriate Douglas Process Standard (DPS) will be carefully followed. Standard procedures for the use of oversize bolts and rivets and stepped enlargement to the next standard size is established by DPS and this procedure will be followed as necessary.

The all metal constant section box is shown in Figure 20 in the subassembly stages. Final assembly of the completed stabilizer involves the match machining of the two outboard panel box sections to the constant section center box which is done by the special milling fixture to establish flight alignment. This fixture is shown in Figure 21. After the operating bulkheads are assembled on both the left and right structural box assemblies, the bulkheads are faced (machined) for alignment then match drilled on master tooling. The holes are ream fitted and identified in the mating condition which makes them non-interchangeable. Various testing instrumentation will be installed in inaccessible areas during assembly. The manufacturing flow diagram is shown for both center section and outer panels (Figures 22 and 23). Exploded views of the constant section center box and the outboard box showing the composite material bonded in place are shown in Figures 24 and 25.

After completion of the structural section, the leading edge and tips will be removed from the used assembly and prepared for reuse. Attach holes will be back drilled and the leading edge and tips bolted on.

3.3.4 Horizontal Stabilizer Installation

The possible field installation will be done at an airline overhaul facility with an installation crew sent to the area for the change. The completed stabilizer will be removed from its shipping crate and installed per the procedure outlined in Appendix F.

The metal stabilizer will be stored in the shipping crate and the elevators reinstalled on the aircraft. Flight controls will be reconnected and all test instrumentation will be set up within the aircraft. During the vibration tests the crew will assist in the rigging of the aircraft and any internal access checks required. Upon completion of the flight service period a crew will be sent to restore the aircraft to its original condition and store the hybrid composite assembly or otherwise dispose of it.

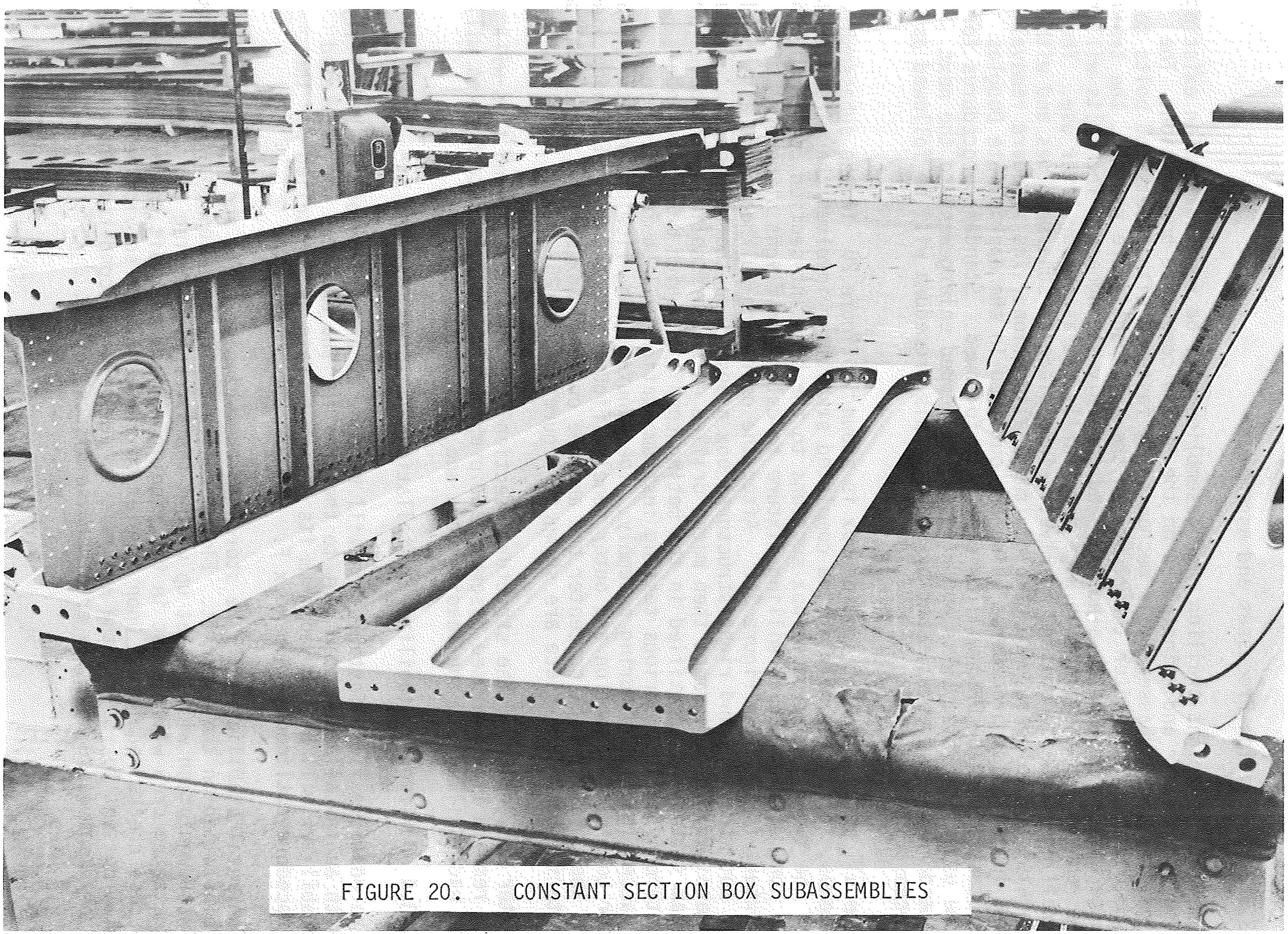


FIGURE 20. CONSTANT SECTION BOX SUBASSEMBLIES

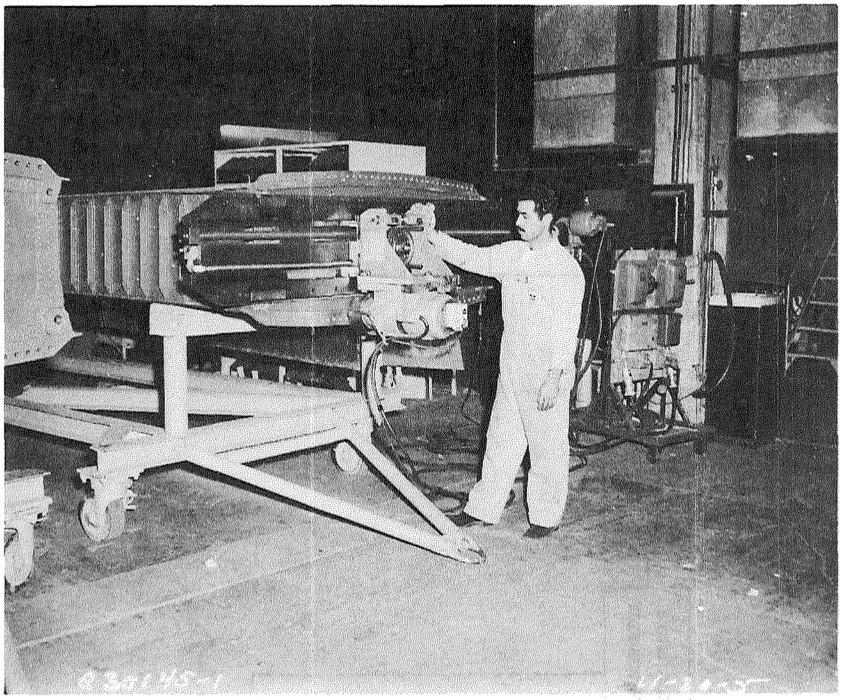


FIGURE 21. MILLING FIXTURE FOR MACHINING OUTBOARD SECTION CENTER SECTION MATING PLANE

3.4 TESTING

A ground test program has been defined to verify the design and structural integrity of the hybrid concept and experimentally determine its stiffness and static strength characteristics.

The test program will be divided into two phases, (a) Development and (b) Certification. These phases will be inter-related and, for the most part, will occur concurrently. The program will be of an evolutionary nature, commencing with the Data Generation Test and culminating with the Horizontal Stabilizer Tests. The chronological sequence of testing is shown in Figure 26. It is anticipated that the data acquired from each group of tests will be utilized in the subsequent design of specimens, eliminating unnecessary duplication of tests and providing an economical and efficient growth within the program. The test effort can be appropriately divided into three groups:

- o Static and Fatigue Data Generation and Design Allowable Tests
- o Static and Fatigue Subcomponent Tests
- o Horizontal Stabilizer Tests

Successful achievement of the objectives in these tests will establish the flight worthiness of the design concept.

The following test setups, instrumentation and descriptions are anticipated for response to the program, but are subject to revision and finalization during the initial portion of the program after final selection of concepts. Prepreg materials and adhesives which are new at the start of Phase II and not fully characterized but show a potential benefit for the program will be incorporated into the test plan.

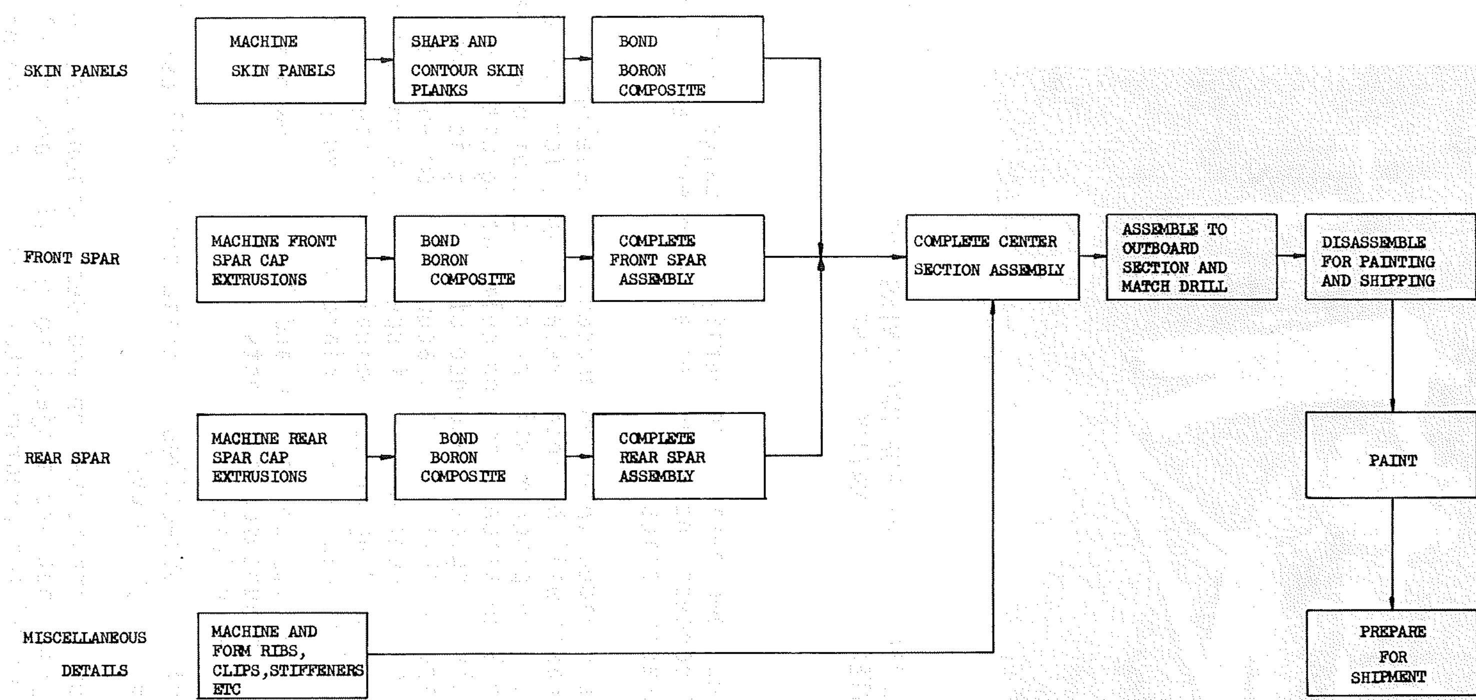


FIGURE 22. MANUFACTURING FLOW PLAN FOR CONSTANT SECTION

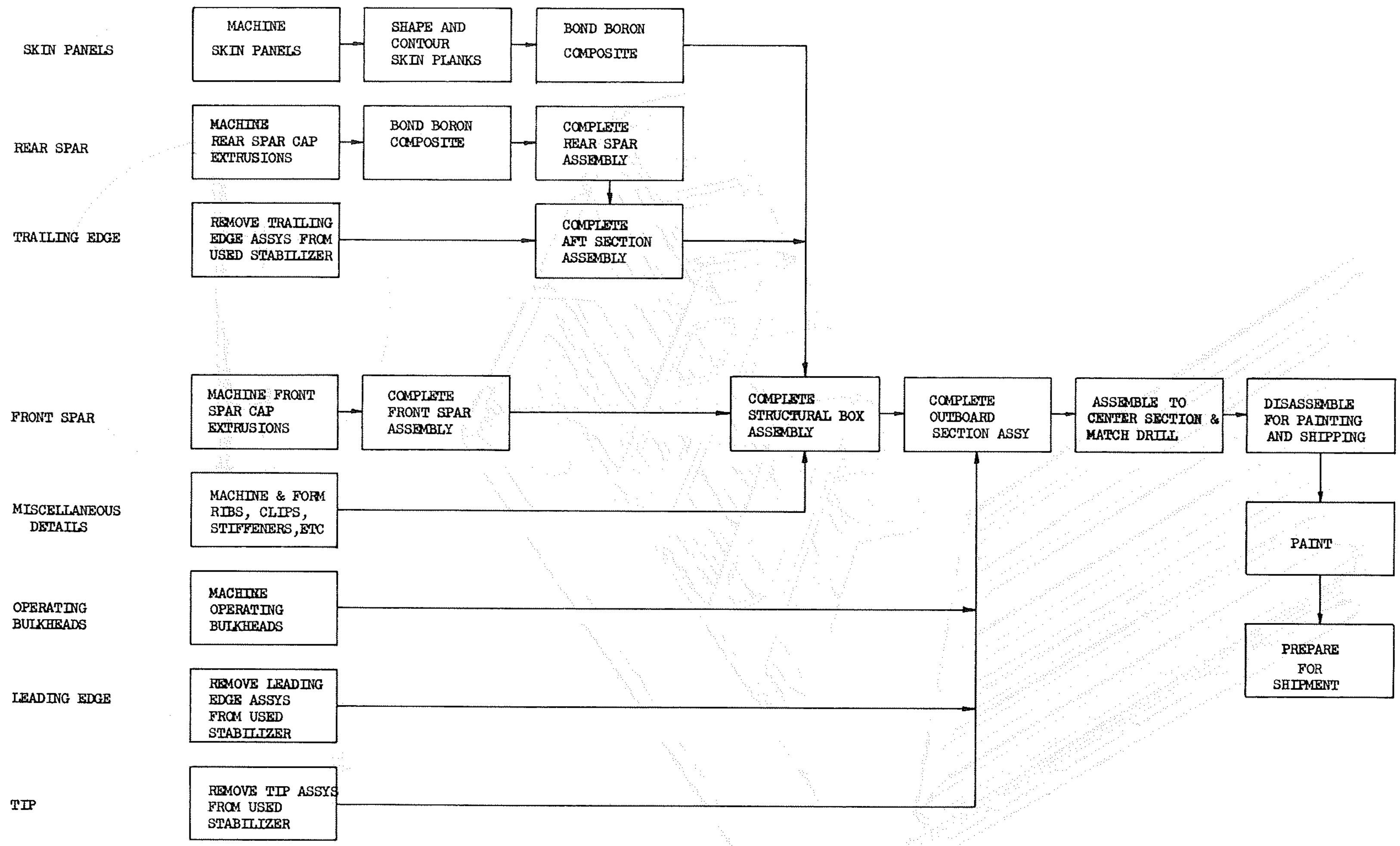


FIGURE 23. MANUFACTURING FLOW PLAN FOR OUTBOARD SECTIONS

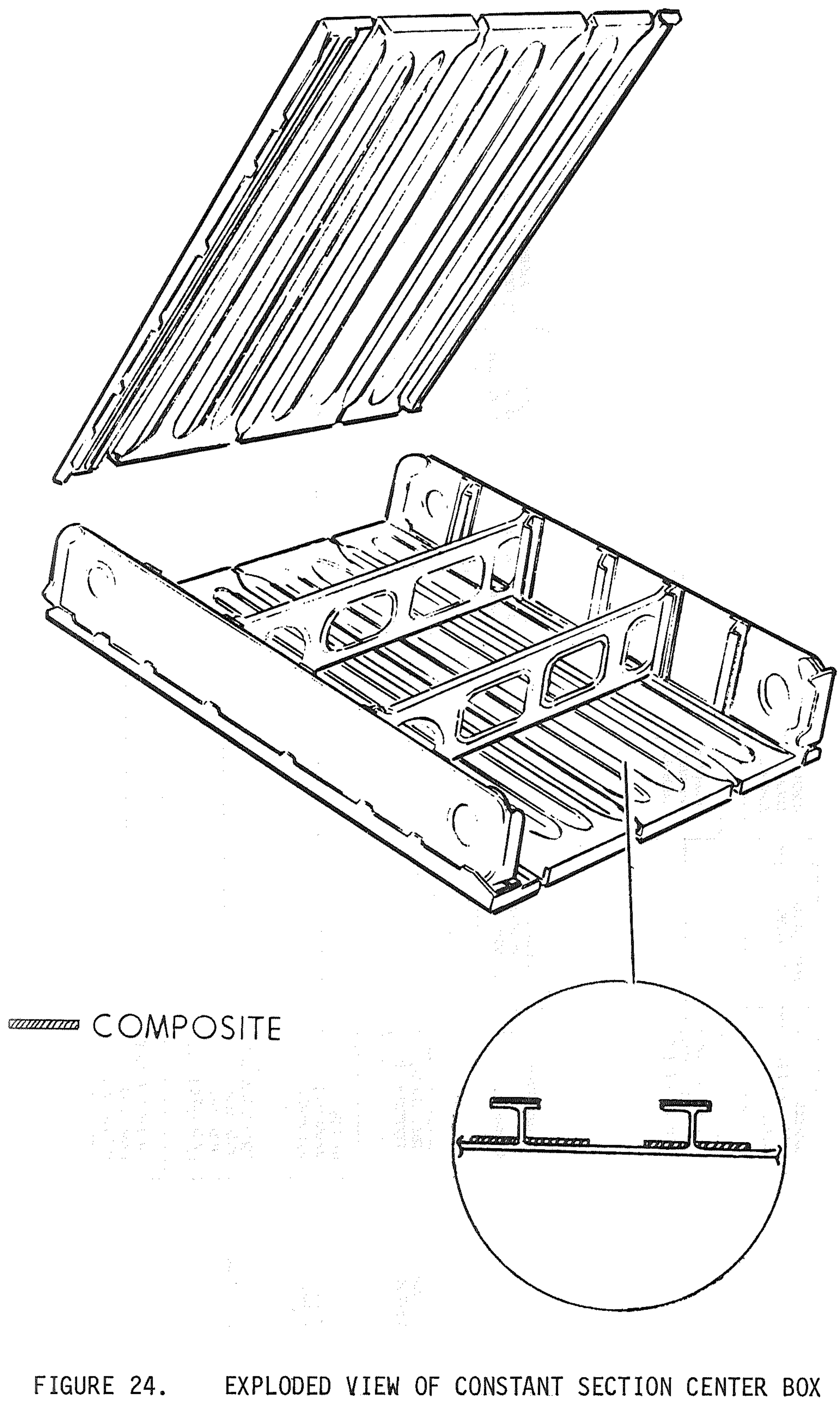


FIGURE 24. EXPLODED VIEW OF CONSTANT SECTION CENTER BOX

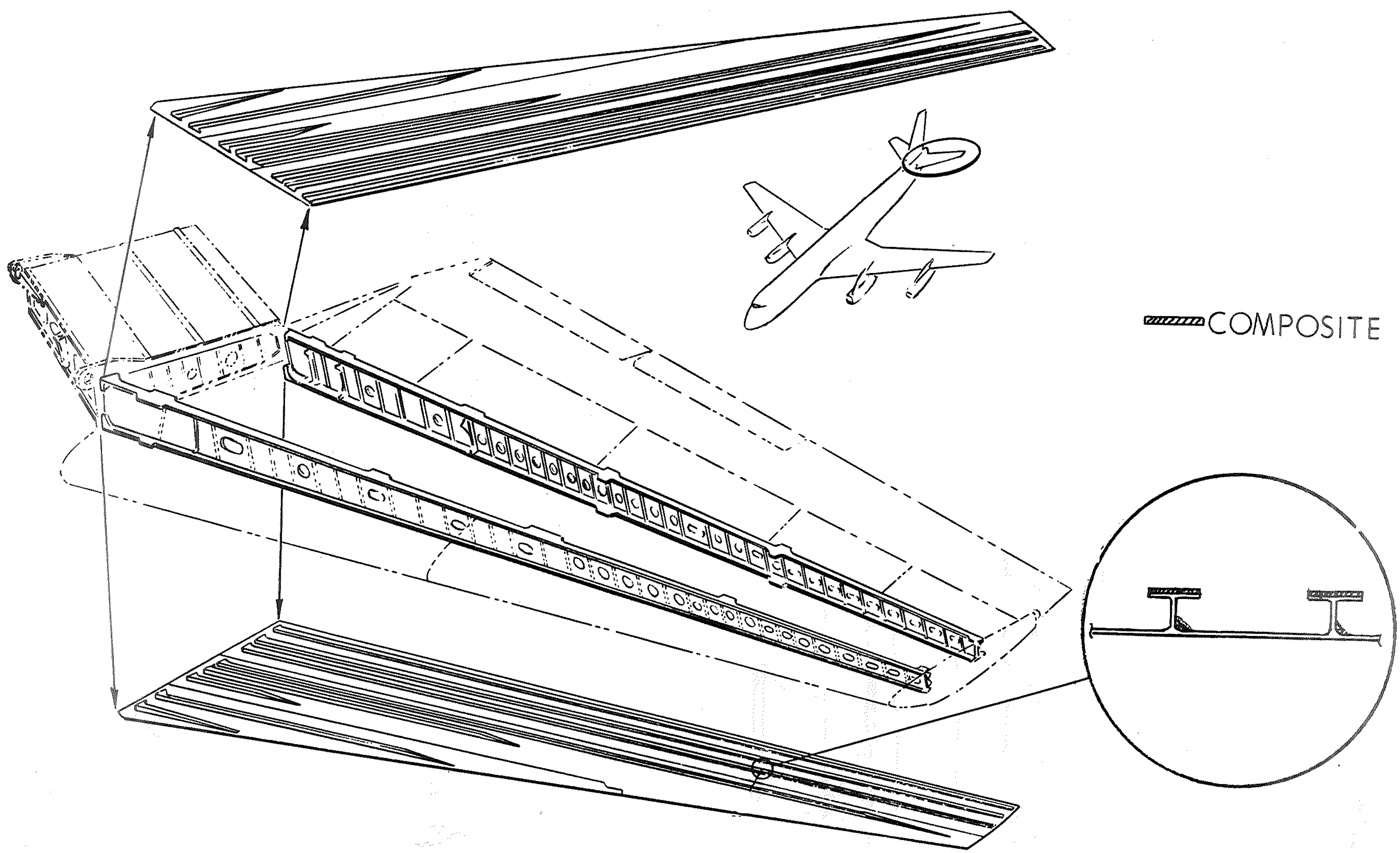


FIGURE 25.   EXPLODED VIEW OF OUTBOARD STRUCTURAL BOX

3.4.1 Design Data Tests

The design data tests will provide information for establishing design allowables, substantiating the processing and bonding techniques, and verifying the design concept. The tests are described herein and summarized in Table II.

Static test data, fundamental to the program, will be generated from tests performed to failure on a Universal testing machine. Loads will be applied at the rate of 0.05 inch per minute. Usually the load will be applied through serrated machine grips but pin-loading will also be used as shown in Table II. Appropriate lateral support will be provided to stabilize the compression specimens. The gripping tabs on each specimen will be checked to ensure that they are flat and parallel prior to testing.

Fatigue data will be generated from test specimens subjected to a constant amplitude fatigue environment with loads being applied at the rate of 1800 cycles per minute. Cycling will continue to failure or runout (no failure at 10^6 cycles), whichever occurs first. All runout specimens will then be tested for residual static strength. Maximum and minimum stress levels will be selected for each specimen based on analysis and on initial test results.

<u>Laminate Tests</u>

Because the composite is to be loaded only in tension or compression, it will not be tested under in-plane shear. Tension and compression tests for the composite will be performed yielding complete stress-strain records. The influence of temperature throughout the operating range will be extrapolated from tensile measurements alone. The laminate interlaminar shear test data is required for the detail joint designs. Fatigue loading in interlaminar shear will be used as the basis for selecting the best of two resin systems since the resin matrix is loaded most severely in the joints at each end of the laminate strips.

<u>Adhesive Shear Tests</u>

The basis for the design of the adhesive-bonded joints is the complete stress-strain curve for an adhesive film in shear. A preliminary screening test for six adhesives will be the peel tests, at three temperatures throughout the operating range. The symmetric specimen chosen is such as to remove any tendency for the filamentary composite to bend and split and thereby transfer the failure surface out of the bond. The surface preparation to be used will be based on the best available test data and experience. Peel testing is also a qualitative indication of the ductility of adhesives. The four best adhesives identified by the peel tests above will be further examined to produce complete stress-strain records in shear at three temperatures throughout the operating range. The adherends used will be one-half inch thick aluminum, and the standard chromic-acid etch will be used to prepare the bonding surfaces. The best three adhesives will be used to fabricate hybrid double-lap joint tests in order to ensure compatibility of the adhesive and composite.

TABLE II

SUMMARY OF DESIGN DATA TEST SPECIMENS

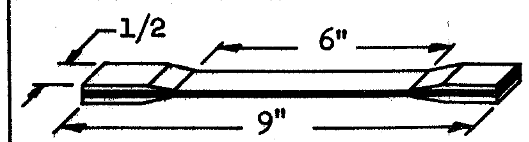
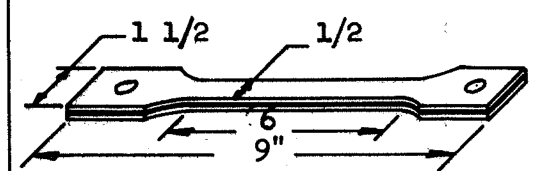
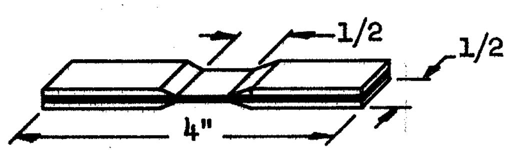
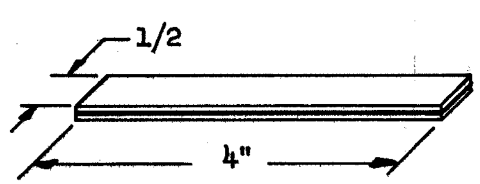
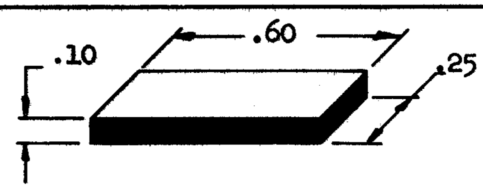
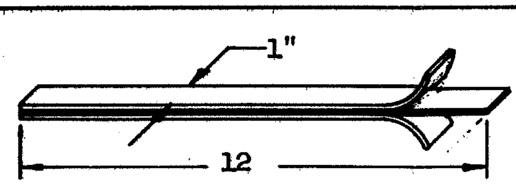
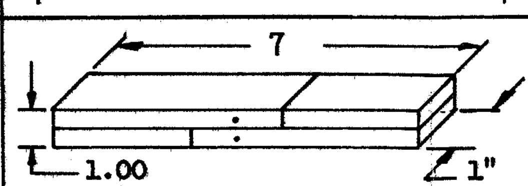
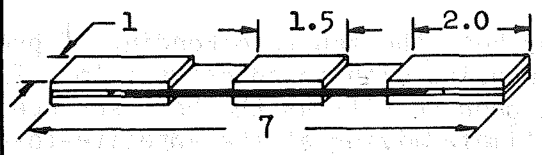
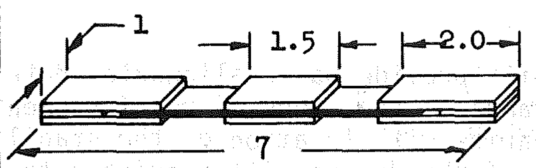
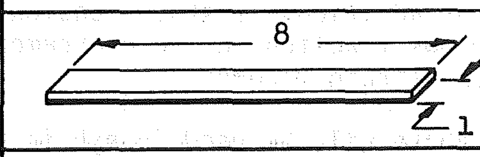
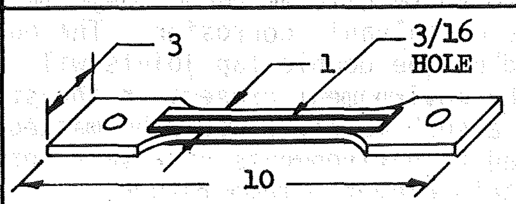
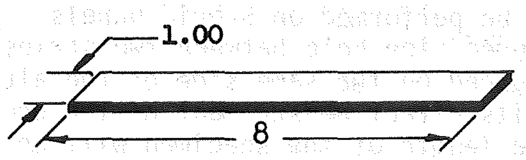
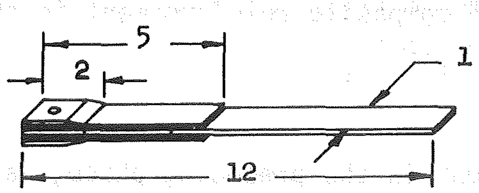
TYPE OF TEST	SPECIMEN CONFIGURATION	TEST OBJECTIVE	FABRICATION DETAILS	VARIABLES	REPLICATES	TOTAL	INSTRUMENTATION
TENSION (LAMINATE)		TO OBTAIN BASIC PROPERTIES E, σ, ε	UNIDIRECTIONAL (0°) .040 THICK	3 TEMPERATURES RT, -67°, 180°	7	21	EXTENSOMETER THERMOCOUPLE
TENSION (HYBRID)		TO OBTAIN BASIC PROPERTIES E, σ, ε	BORON (0°) ALUMINUM 1 .040 .020 2 .040 .040 3 .040 .100 4 .040 .200	3 TEMPERATURES 4 THICKNESSES	7	84	EXTENSOMETER THERMOCOUPLE
COMPRESSION (LAMINATE)		TO OBTAIN BASIC PROPERTIES E, σ, ε	UNIDIRECTIONAL (0°) .040 THICK	3 TEMPERATURES	7	21	STRAIN GAGE THERMOCOUPLE
COMPRESSION (HYBRID)		TO OBTAIN BASIC PROPERTIES E, σ, ε	BORON (0°) ALUMINUM 1 .01 .05 2 .02 .05 3 .04 .04 4 .06 .03 5 .04 .20	3 TEMPERATURES 5 THICKNESSES	7	105	STRAIN GAGE THERMOCOUPLE
INTERLAMINAR SHEAR		TO OBTAIN BASIC PROPERTIES τ	UNIDIRECTIONAL (0°) .10 THICK	3 TEMPERATURES	7	21	THERMOCOUPLE
PEEL		TO OBTAIN PEEL STRENGTH OF ADHESIVE	.020 Al .040 BORON (0°)	3 TEMPERATURES 6 ADHESIVES	3	54	THERMOCOUPLE
TENSION LAP SHEAR		OBTAIN ADHESIVE STRESS-STRAIN DATA	.50 ALUMINUM PLATE	4 ADHESIVE SYSTEMS 3 TEMPERATURES	7	84	EXTENSOMETER THERMOCOUPLE

TABLE II

SUMMARY OF DESIGN DATA TEST SPECIMENS

TYPE OF TEST	SPECIMEN CONFIGURATION	TEST OBJECTIVE	FABRICATION DETAILS	VARIABLES	REPLICATES	TOTAL	INSTRUMENTATION
TENSION DOUBLE LAP		TO DETERMINE THE COMPATIBILITY OF ADHESIVE	.060 BORON .063 ALUMINUM	3 ADHESIVE SYS	5	15	NONE
TENSION DOUBLE LAP		TO DETERMINE SATISFACTORY SURFACE PREPARATION FOR ADHESIVE	.060 BORON .063 ALUMINUM	2 ADHESIVE SYS 1-Grit Blasting 2-Nylon Peel Ply 3-Scuff Sanding	7	42	NONE
CURVATURE MEASUREMENT		DETERMINE STRESS FREE TEMPERATURE FOR ADHESIVE	.063 ALUMINUM .030 BORON (0^{o}) REQUIRES RIGID FLAT CURING FIXTURE	3 ADHESIVES 2 CURE CYCLES	2	12	NONE
TENSION-TENSION FATIGUE		OBTAIN FATIGUE DATA ON HYBRID COMPOSITE	.080 ALUMINUM .025 BORON (0^{o}) TEST METAL CUT FROM DIFFERENT SECTIONS OF SHEET	2 "R" VALUES	5	10	NONE
3-POINT FLEXURE FATIGUE		TO VERIFY THE ADEQUACY OF THE MATRIX INTERLAMINAR SHEAR STRENGTH	.10 BORON (0^{o})	2 RESIN SYSTEMS 2 "R" VALUES	3	12	NONE
GALVANIC CORROSION		TO DETERMINE GALVANIC CORROSION EFFECT	.030 BORON (0^{o}) .180 ALUMINUM FABRICATE 3 SETS OF SPECIMENS WITH 2 DIFFERENT ADHESIVES	2 ADHESIVES 2 STRESS LEVEL FOR ONE OF THE ADHESIVES 2 EXPOSURE	5	30	NONE

Finally, tests will be run on the two most promising adhesives to ensure that a surface preparation is being used which is adequate to ensure the prevention of failure by inadequate adhesion to the composite. An economical test coupon incorporating load tabs bonded to the center and a tab and spacer arrangement bonded to each end, will be used as shown in Table III, to yield two data points from each specimen fabricated.

The proposed design calls for both the separate bonding of pre-cured composite parts and for co-curing and bonding with an adhesive film. The latter process can be predicted to yield as good results as the best surface preparation for the former because of the intimate mixing at the adhesive-to-composite interface.

Hybrid Composite Tests

The hybrid composite tests will provide basic allowables data for the design. It is necessary to cover a range of fractions of reinforcement for two reasons. The reinforcement fraction varies with location on the stabilizer and, minor changes during the evolution of the design will require a test data basis without holding up the program for additional tests. Tension and compression tests for hybrids containing various fractions of reinforcement will be performed yielding complete stress-strain records.

The two best adhesives selected above will be used in hybrid bonded joints designed to assess the effects of galvanic corrosion. The outer adherends, rather than the inner adherend of the double-lap joints will be made of composite to simulate the exposure to the environment present in the structure. Otherwise any tendency of the matrix to absorb moisture would be masked. The specimens will be preloaded and subjected to environments of a salt spray and of a high relative humidity at a moderately elevated temperature.

Crack propagation tests will be performed on hybrid panels. Each panel will contain a centrally located precision hole between two strips of longitudinal boron-epoxy composite both bonded on the same side of the aluminum. This unbalanced construction permits direct measurement of the growth of the crack. Transverse supports alone the length of the specimen will be provided during the tests to eliminate the effects of the eccentric load path. The spreading of a debond around the crack will be observed and recorded, because of the proximity of drain holes to the composite reinforcement in the constant section of the horizontal stabilizer.

3.4.2 Subcomponent Tests

Using the ground work development in the preceding phase, data will be generated which are directly related to the substantiation of the hybrid design concept. The tests outlined in Table III will encompass three basic areas: (1) the bonded scarf joint at the inboard ends of the composite reinforcement, (2) crippling of the hybrid spar caps, and (3) the behavior in compression and in shear of the stiffened skin panels.

TABLE III

SUMMARY OF SUBCOMPONENT TEST SPECIMENS

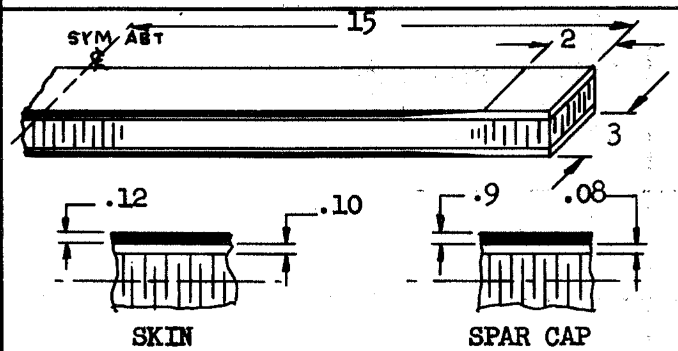
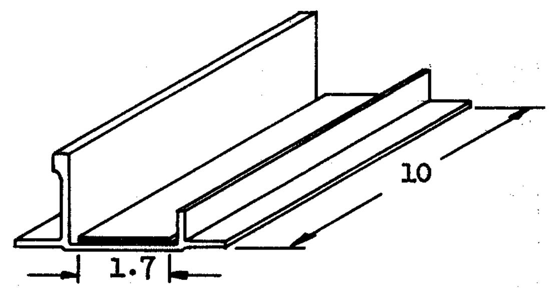
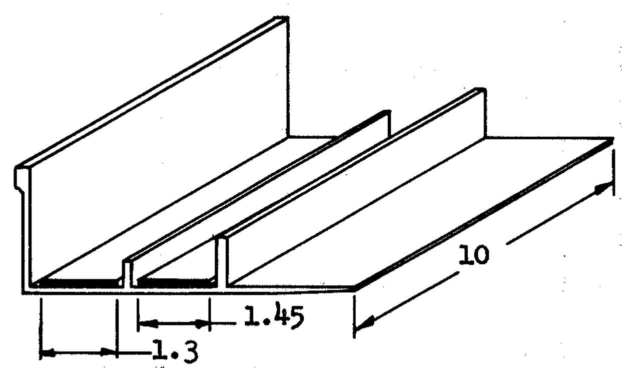
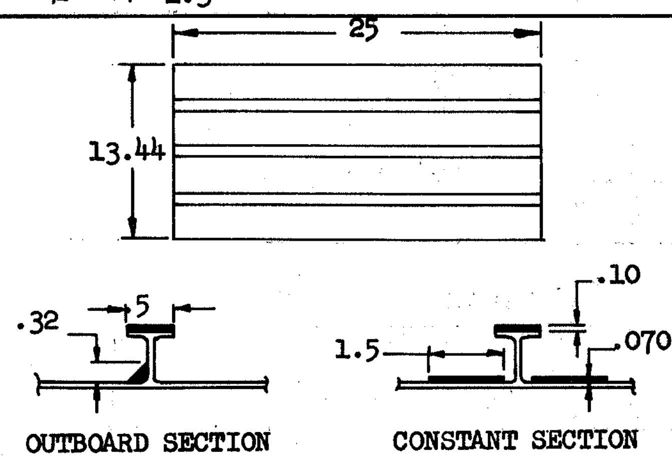
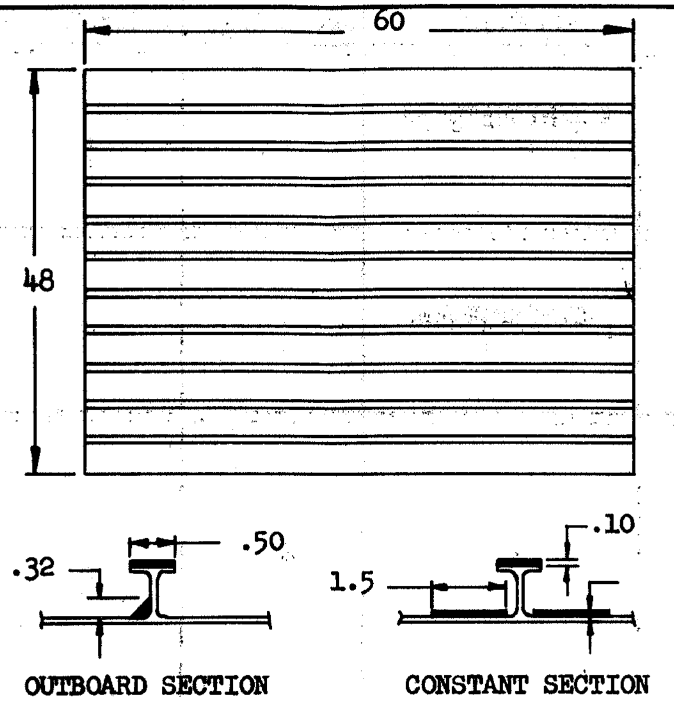
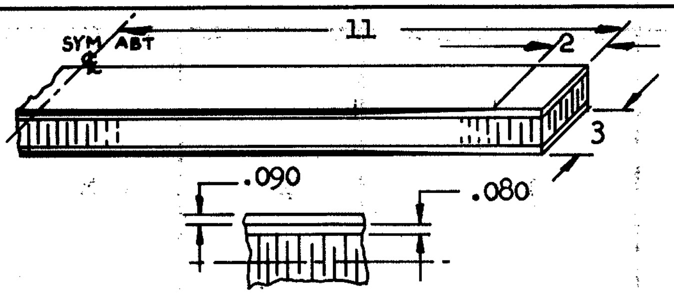
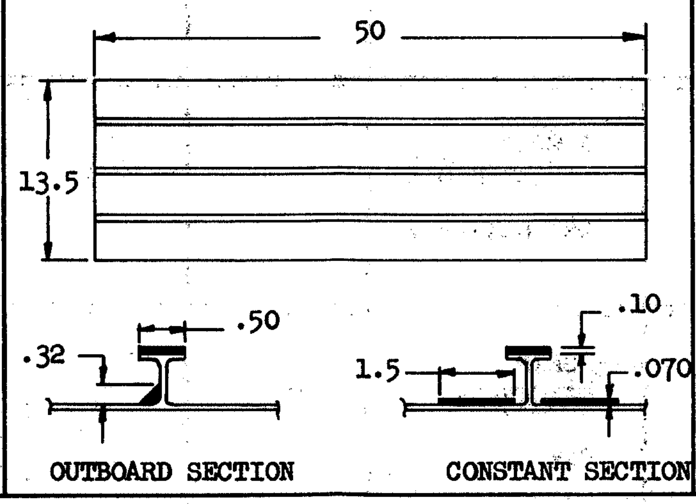
TYPE OF TEST	SPECIMEN CONFIGURATION	TEST OBJECTIVE	FABRICATION DETAILS	VARIABLES	REPLICATES	TOTAL	INSTRUMENTATION
COMPRESSION		TO DETERMINE COMPOSITE TO METAL TAPERED JOINT LOAD TRANSFER CAPABILITY	INBD END - SKIN .12 BORON (0°) .10 ALUMINUM	3 TEMPERATURES	1	3	3 STRAIN GAGES 1 DEFLECTION GAGE
			INBD END - SPAR CAP .09 BORON (0°) .08 ALUMINUM TAPER BUILDUP AS SHOWN	3 TEMPERATURES	3	9	
COMPRESSION		OBTAIN COMPRESSIVE CRIPPLING STRENGTH DATA ON OUTBOARD REAR SPAR CAP	1.7 x .130 x 10.0 BORON (0°)	1	2	2	3 STRAIN GAGES 1 DEFLECTION GAGE
COMPRESSION		OBTAIN COMPRESSIVE CRIPPLING DATA ON CONSTANT REAR SPAR CAP	1.3 x .10 x 10.0 BORON 1.45 x .10 x 10.0 BORON ALL (0°)	1	2	2	3 STRAIN GAGES 1 DEFLECTION GAGE
COMPRESSION PANEL		TO VERIFY COMPRESSIVE STRENGTH AND TO EVALUATE FABRICATION PROBLEMS OF STIFFENED HYBRID PANELS	OUTBOARD SECTION ALL UNIDIRECTIONAL BORON COMPOSITE	2 SKIN THICKNESSES .045 and .070	2	4	10 STRAIN GAGES
			CONSTANT SECTION ALL UNIDIRECTIONAL BORON COMPOSITE	1 SKIN THICKNESS .045	2	2	2 DEFLECTION GAGES

TABLE III

SUMMARY OF SUBCOMPONENT TEST SPECIMENS

TYPE OF TEST	SPECIMEN CONFIGURATION	TEST OBJECTIVE	FABRICATION DETAILS	VARIABLES	REPLICATES	TOTAL	INSTRUMENTATION
SHEAR PANEL		VERIFY SHEAR STRENGTH OF STIFFENED	OUTBOARD SECTION ALL UNIDIRECTIONAL BORON COMPOSITE	2 SKIN THICKNESSES .045 & .070	2	4	30 STRAIN GAGES
			CONSTANT SECTION ALL UNIDIRECTIONAL BORON COMPOSITE	1 SKIN THICKNESS .045	2	2	
FATIGUE COMP-COMP		DETERMINE FATIGUE LIFE (CONSTANT AMPLITUDE)	.090 BORON (0°) .08 ALUMINUM	1 "R" VALUE	3	3	NONE
FATIGUE SHEAR PANEL		TO VERIFY FATIGUE LIFE IN SHEAR	OUTBOARD SECTION ALL UNIDIRECTIONAL BORON COMPOSITE	2 SKIN THICKNESSES .045 AND .070	2	4	20 STRAIN GAGES
			CONSTANT SECTION ALL UNIDIRECTIONAL BORON COMPOSITE	1 SKIN THICKNESS .045	2	2	

Bonded Scarf Joints in Hybrid Composite

The scarfed joints will be designed on the basis of the material properties data generated during Phase II. For certification purposes it will be necessary to demonstrate the adequacy of the joint design. Static tests will be performed on the joints for both the spar cap and the skin stiffeners. The nature of the residual bond stresses is such that they add to those due to an external compression load and subtract from, or alleviate, those due to an external tensile load. Since the joints would be exposed to both tensile and compressive loads in service, they will be tested in compression which is the critical case. Although tension tests would be easier to perform the results would provide only an unduly optimistic assessment of the joint strength. The tests will be conducted at three temperatures throughout the service regime because of the varying adhesive properties and residual thermal stresses.

The honeycomb shown in the specimen illustrated in Table III is to stabilize the joints being tested. Static tests will be conducted on a Universal testing machine with load being applied at the rate of 0.05 inch per minute. Because of the high residual thermal stresses it is considered necessary to test one of the joints under fatigue loading. Fatigue loading will be to a constant amplitude at a rate of 1800 cycles per minute with an R value (minimum load/maximum load) determined by an analysis of the flight loads and residual stresses for the proposed design. Cycling will continue to failure or runout (no failure at 10^6 cycles), whichever occurs first. All runout specimens will be tested for residual static strength.

Hybrid Composite Spar-Cap Crippling Tests

Crippling data will be obtained from compression tests of uniform hybrid composite sections representative of the spar cap sections. The specimens will be machined flat and parallel on each end to ensure that the load applied through stiff loading plates is introduced uniformly into both the filamentary composite and the metal. Loads will be applied by a Universal testing machine at the rate of 0.05 inch per minute until failure develops.

Integrally-Stiffened Skin Compression Panels

Each test panel simulates a one-bay span between ribs and has the width of one of three such panels on the structure. The critical panel is the central one which is effectively supported at its ends. The test panels will be mounted in an end support system consisting of Vee blocks and Vee grooves with the sides free in order to experimentally verify the compressive strength of the stiffened hybrid panels. A compressive load at the ends will be applied by a Universal testing machine at a rate of 0.05 inch per minute. Since a buckling instability is sought, a Southwell plot will be prepared to establish the strength.

Integrally-Stiffened Skin Shear Panels

Static and fatigue tests will be performed to verify the shear strength of the stiffened hybrid panels. The skin is designed to operate as a tension field shear panel prior to ultimate load, but beyond the frequently occurring flight loads. The fatigue tests are necessary to demonstrate adequate integrity of the boron-epoxy reinforcement, and of the bond adjacent to the skin under the wrinkling action of the skin between the stiffeners. Each specimen will be installed in a large capacity picture-frame test fixture designed specifically for testing shear panels. The fixture consists of a rigid-side, corner-hinged adjustable-size steel fixture that introduces a pure shear loading to the test panel mounted in the frame. The picture-frame is pinned to a rigid support at one end, while a hydraulic loading actuator applies the test load at the other end. The test panel is mounted inside of the frame by bolting to an intermediate hat-section that permits a certain amount of breathing-action to minimize compression stresses while applying the shear load. A photograph of a test apparatus with a shear panel mounted in the frame ready for test is shown in Figure 27.

3.4.3 Horizontal Stabilizer Component Tests

The ground test program will culminate in a series of tests designed to conclusively establish the flight worthiness of the hybrid composite horizontal stabilizer concept. The loads and load conditions will be identical to those applied when qualifying the original DC-8 horizontal stabilizer, consequently a direct comparison of the test data from these two programs can be made.

The end-item test article consisting of the complete horizontal stabilizer structure (two outboard structural assemblies and the constant section center box) plus the control surfaces will include all of the hybrid composite details. In addition, the elevator, elevator flying tab, and elevator geared tab control systems along with the horizontal stabilizer electro-hydraulic actuator control system will be provided. These systems will be employed for placing each surface in the proper position prior to load application. The stabilizer control system will also be utilized for operating the stabilizer while external loads are applied. The elevators, elevator flying tabs, and elevator geared tabs will not be actuated during the tests. Consequently the elevator actuating rods which are attached to the elevator torque tubes will be replaced by rod and calibrated load cell assemblies to insure that the elevator positions remain fixed, thereby fixing the positions of the elevator flying tabs and elevator geared tabs.

The test conditions have been selected by the detailed survey of the horizontal stabilizer external loads and internal loads. These conditions produce critical design loads for the principal structural elements and maximum loads for the majority of the horizontal stabilizer structures system. A summary of the critical design loads and test conditions appears in Table IV.

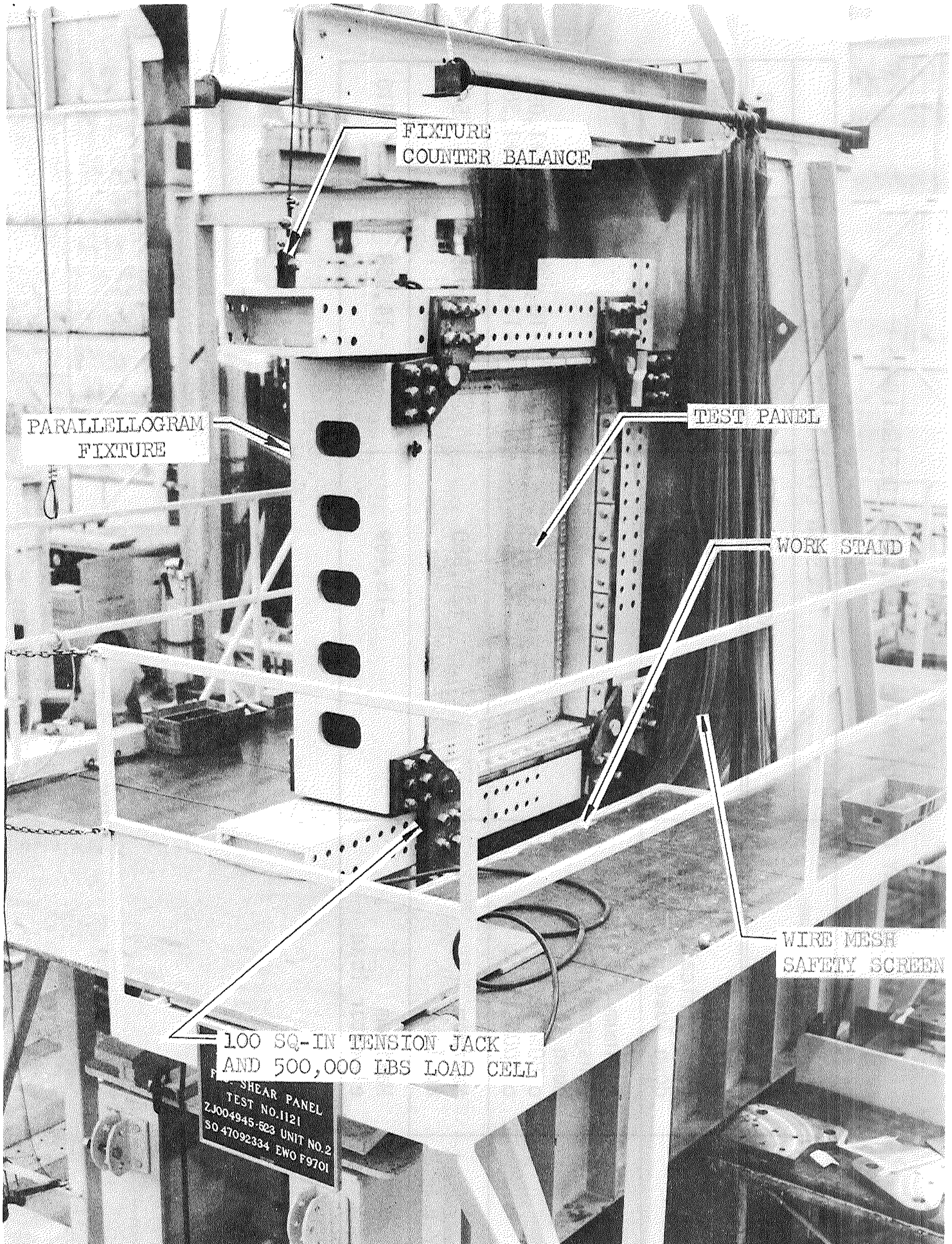


FIGURE 27. SHEAR PANEL TEST SETUP

TABLE IV

SUMMARY OF HORIZONTAL TAIL CRITICAL DESIGN LOADS

TEST CONDITION NUMBER	TYPE OF LOADING ON TAIL	DESIGN LIMIT LOADS PER SIDE (LB)		
		STABILIZER	ELEVATOR	TAB
1	Maximum (symmetrical) down load	-22,955	-13,100	+1,365
1A	Maximum (unsymmetrical) down load o 80% left o 100% right	 -18,364 -22,955	 -10,480 -13,100	 +1,092 +1,365
2	Maximum up load	+21,980	+ 4,035	- 545
3	Maximum elevator load	-15,489	-14,958	+ 810

The horizontal stabilizer will be installed in a structural test loading fixture with rigid supports for the stabilizer attach points (stabilizer-fuselage hinge and actuator fittings). A typical test setup is shown in Figure 28. This loading fixture will also include a support structure for mounting in the proper configuration the horizontal stabilizer and elevator control systems, and will provide the necessary flexibility to accommodate various tests conditions, each of which will require repositioning of the hydraulic loading actuators.

Test loads will be applied to the horizontal stabilizer and elevators through a loading system consisting of hydraulic actuators in conjunction with various whiffletree arrangements, each of which will distribute loads to pads attached along the specimen surfaces. These pads will be the mechanism for subsequently transferring the loads to the specimen. Spanwise and chordwise loads will be applied by setting the hydraulic actuators at the proper angles with respect to the test article.

Instrumentation of the specimen will consist primarily of bonded electrical-resistance strain gages and position transducers. The gages and transducers will be positioned at critical load points as indicated by analysis and results of earlier testing and will provide complete load-strain and stiffness response of the stabilizer. Load, strain and displacement data will be recorded on magnetic tape by the Douglas high speed data gathering system.

These test data will be reduced to a form adaptable for further studies and for correlation with design and analytical data. A precise and thorough study of all data will be performed, supplemented by an extensive examination of the test article after each load condition to establish the extent of damage, if any, and mode of degradation. The test data will be presented, where possible, in both tabulated and graphical form to facilitate interpretation.

Ground Vibration Test

A two day ground vibration test will be conducted prior to first flight. The aircraft will be configurated in flight condition with the hybrid horizontal stabilizer installed and with a production set of DC-8 elevators. Frequency surveys will be conducted using electromagnetic shakers and the important resonant frequencies and mode shapes established. Structural damping of the important modes will also be determined. The modal data will be plotted and compared to existing DC-8 and theoretical modal data for partial confirmation of flutter stability. In addition, the measured mode shapes and frequencies will be input to flutter computing programs to further establish stability margins prior to first flight.

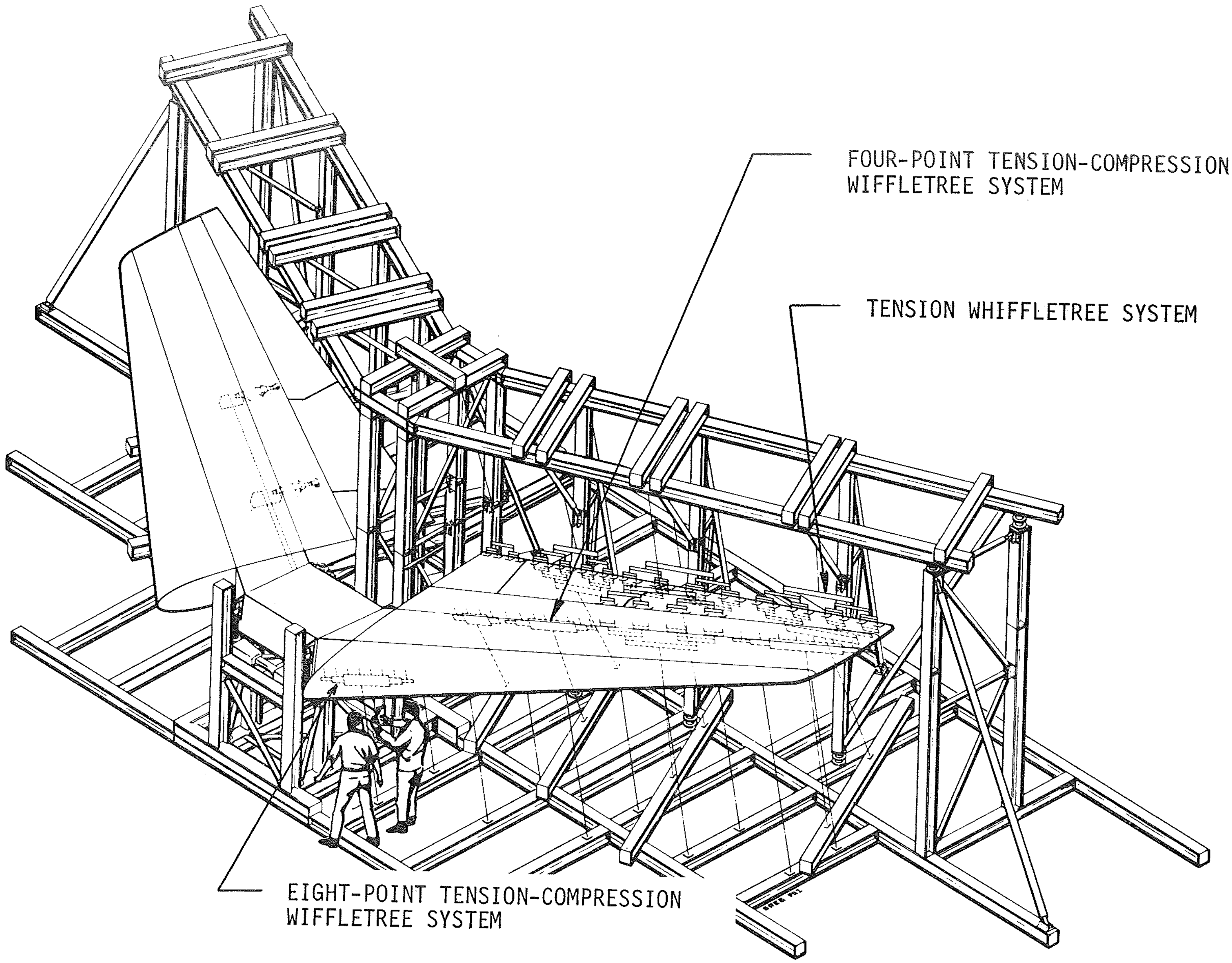


FIGURE 28. TYPICAL HORIZONTAL STABILIZER PROOF TEST SETUP

Fatigue Test

A fatigue test will be conducted on the ground test article (Recommended Demonstration Program only). The total test on the horizontal stabilizer will include the effects of gusts, ground-air-ground cycles, basic conditions and landing impact (Figure 29). The actual complex fatigue loading will be reduced to a simple but representative and meaningful test spectrum for fatigue through a series of typical flight spectra applied in the appropriate number of loading blocks. The low temperature operating environment during the long exposure at cruise above 30,000 feet is such as to affect significantly the thermally-induced locked-in stresses. It is planned to compensate for this factor during testing at room temperature by increasing the applied loads to represent the resultant loads on the critical upper (tension) skin and lower (compression) bonded scarf joints at the ends of the reinforcement.

3.5 QUALITY ASSURANCE

3.5.1 Quality Assurance Approach

Process control and nondestructive test (NDT) activities shall be performed continuously on the program from initiation to completion. These activities will fall into four main categories: (1) Accountability, (2) Traceability, (3) Documentation, and (4) Quality Control.

Accountability

All critical materials shall be accounted for by weight to provide detailed information on material efficiency and types of loss (e.g., trim, scrap, QC, etc.).

Traceability

Traceability of raw materials shall begin at the level of the supplier to Douglas and shall be so organized as to provide lot and unit identity of all materials used in each final end item. Furthermore, lot and unit definitions shall be so established as to be assignable to a specific manufacturing process or batch.

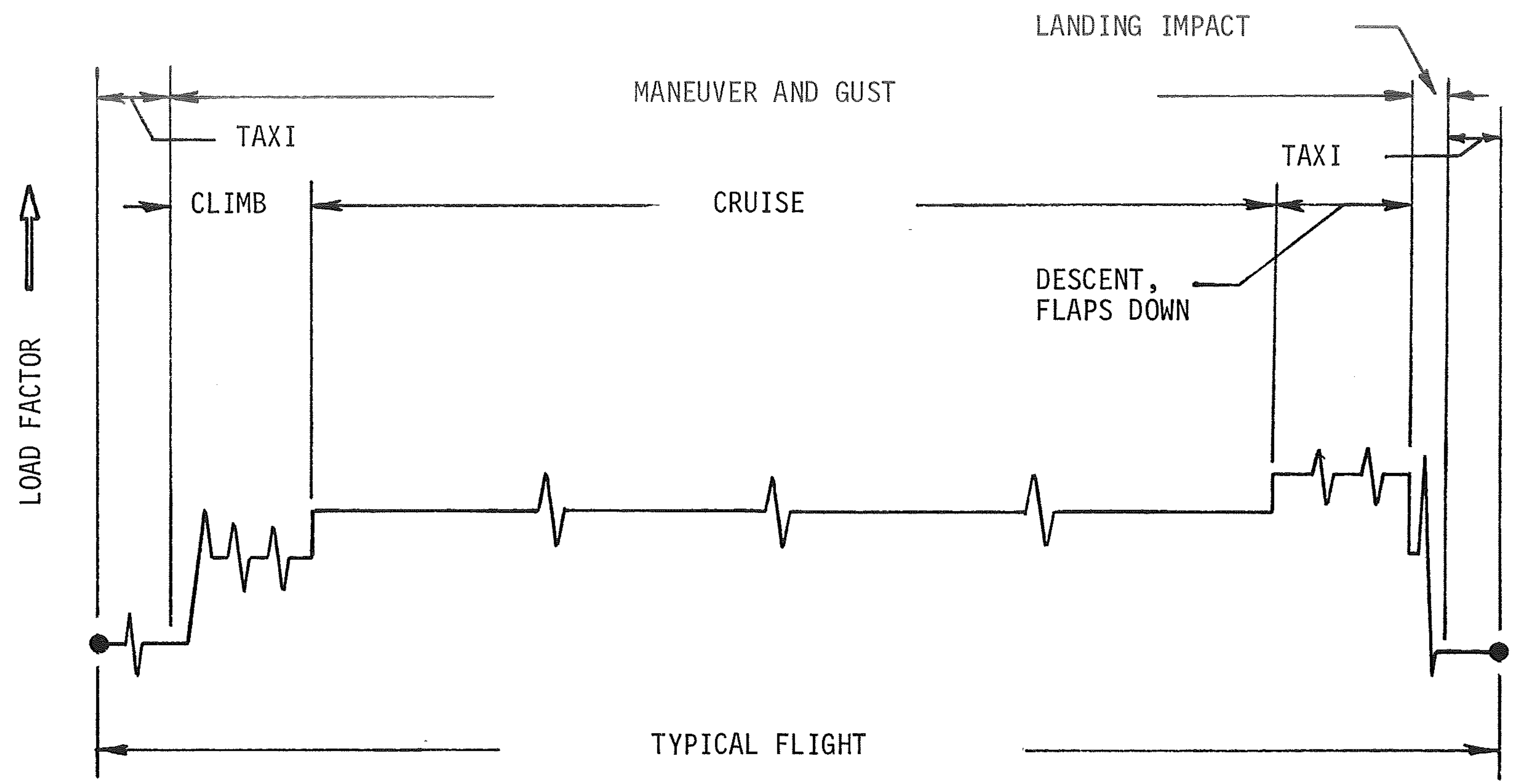
Documentation

Documentation requirements shall be established that will permit:

- Complete identification
- Lot and unit traceability
- Recording of process parameters, specifications and conditions
- Material accountability
- Identification and results of all quality control testing
- Identification and results of all end item testing

3.5.2 Quality Control

The quality control activities shall be sufficiently comprehensive to ensure the use of the highest possible quality of raw material, an optimum fabrication process and end item quality verification. This shall be accomplished by specification and test verification. See Table V.



TOTAL TEST: 120,000 FLIGHT HOURS
43,000 FLIGHTS
473,000 CYCLES

FIGURE 29. PROPOSED FATIGUE TEST LOADS SPECTRUM (FULL SCALE HORIZONTAL STABILIZER)

TABLE V

SUMMARY OF QUALITY CONTROL TEST SPECIMENS
(INCOMING MATERIALS)

TEST	NUMBER OF BATCHES	TESTS PER BATCH	REPLICATES PER TEST	TOTAL SPECIMENS	DOUGLAS MATERIAL SPECIFICATION
BORON/EPOXY:					
PREPREG RESIN CONTENT	4	10	3	120	1919
PREPREG VOLATILE CONTENT	4	5	1	20	1919
PREPREG GEL TIME	4	5	1	20	1919
FLEXURAL STRENGTH	4	1	3	12	1919
SHORT BEAM SHEAR	4	1	3	12	1919
ADHESIVES:					
TENSILE LAP SHEAR	3	1	5	15	Several Specifications

Specification

All raw materials shall be procured to the appropriate Douglas Material Specification with specific written modifications if necessary. Manufacturing processes shall be controlled by a Douglas Process Specification. All testing, both quality control and performance, shall be defined in detail with appropriate documents before initiation of the tests.

Test Verification

Assessment and verification of quality shall be accomplished by test from raw material receipt through the process to and including end item. Incoming materials will be spot checked by testing for critical properties and comparison with supplier data. Test coupons shall be provided and tested for verification of step-by-step in process quality. This will preclude substandard items and eliminate the costs associated with unnecessary rework and process modification.

Quality control tests will be performed on the end items to verify their quality before commitment to expensive final performance testing or flight. Of necessity the majority of end-item quality control tests will be of a nondestructive nature. Composite laminates will be visually inspected for dimensions, kinks, wrinkles, cracks, etc. Pre-cured laminates will be thickness checked before secondary bonding. Laminates bonded and cured in-place will be thickness checked by taking readings before and after bonding. Cured laminates will be radiographed prior to bonding. Bonded components and assemblies will be 100% inspected for bond quality using contact pulse-echo ultrasonic or eddy-sonic (Harmonic bond tester) whichever is considered most appropriate. The inspection will be performed on the upper and lower stringer-cap and rear spar bonds immediately after bonding but prior to assembly. See Figure 30. Reference standards will be required for variations in metal and laminate thickness.

3.6 FAA CERTIFICATION REQUIREMENTS

A preliminary evaluation has been made with respect to the government agencies whose approval would be required to proceed into a flight program with a hybrid composites DC-8 horizontal stabilizer. The Federal Aviation Administration (FAA) is the only government agency whose requirements must be satisfied insofar as flightworthiness is concerned. FAA certification requirements can be divided into three broad categories, namely; (1) design and static strength analysis, (2) tests performed on the ground, and (3) tests which must be performed by flying the airplane. All required analyses, drawings, specifications, and tests (ground and flight) will be formally documented and submitted for FAA approval and certification of the hybrid composite horizontal stabilizer.

3.6.1 Design and Static Strength Analysis

Formal design and static strength analysis reports are required and will be prepared when the design of the hybrid composite horizontal stabilizer has been finalized.

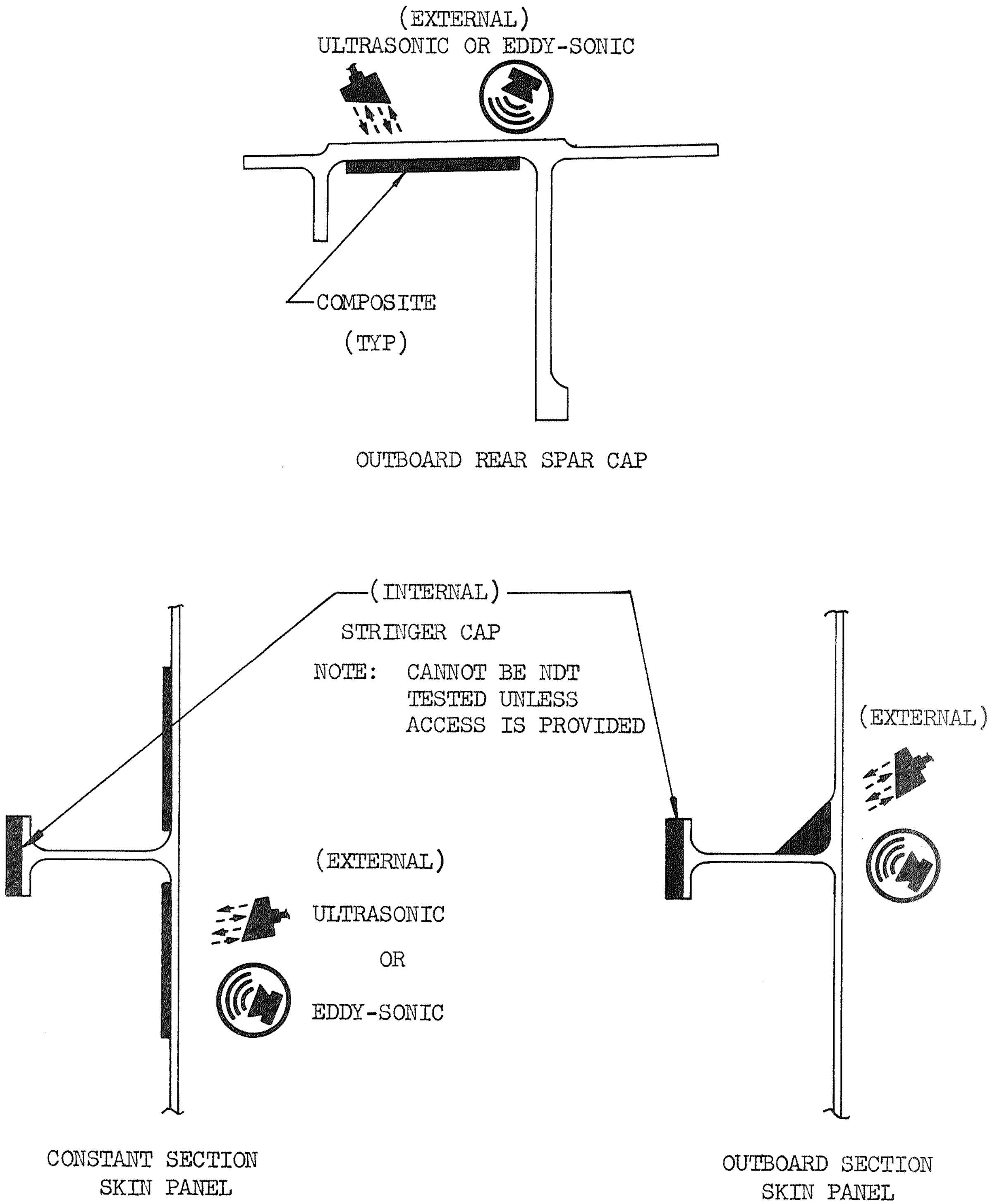


FIGURE 30. IN-SERVICE NDT FOR BOND INTEGRITY

3.6.2 Ground Tests

The ground test portion of the overall certification activities can be divided into specimen tests (design data and subcomponent) and full scale horizontal stabilizer (end-item) tests.

Design Data and Subcomponent Tests

The FAA requires that specimen configurations to be tested and test procedure to be used be approved prior to testing together with verification that completed specimens and test procedure agrees with the drawings and specifications. Static strength and stiffness specimens are required for certification.

Horizontal Stabilizer Tests

A static proof test with the end-item installed in a test fixture is to be conducted with FAA concurrence. A second ground test of the end-item is required with the stabilizer installed on the DC-8 airplane for a ground vibration test (GVT) to verify the analytical computations that adequate safety margins exist from the standpoint of potential flutter and vibration problems.

3.6.3 Flight Certification

Flutter characteristics and elevator control force stability are a function of the stiffness and aeroelastic properties of the horizontal stabilizer. If analytical studies of the aeroelastic characteristics of the hybrid composite horizontal tail show an appreciable change from the basic tail, some flight testing will be required to obtain FAA certification of longitudinal stability and control characteristics. It is anticipated that flight demonstrations of static longitudinal stability in the enroute climb and cruise configurations (CAR-4b, 151, 154, 155) would be required. In addition, it may be necessary to repeat the mistrim controllability demonstration (Special Condition) if the horizontal stabilizer is installed on a Series 60 aircraft.

Flight certification is estimated to require a total of six flights wherein the DC-8 airplane will be evaluated in terms of flutter at critical altitudes and speeds up to design speed/Mach number (V_D/M_D) and longitudinal static stability at critical conditions of airspeed and altitude. Included in the six flights are build-ups to the critical conditions. The specific number of flights required may be modulated (up or down) depending upon the final design and the results of static testing of the end-item and the results of the ground vibration tests.

The required flutter flight testing will consist of flying to V_D/M_D at two altitudes in a sequence of three flights. The FAA requires that the aircraft be designed to have a flutter speed no less than 1.2 V_D. The instrumentation will consist of one accelerometer on each wing tip and horizontal stabilizer tip, a shear strain gage at an inboard horizontal stabilizer spar station, and a position pickup on each elevator. Excitation will be manually induced elevator pulses and oscillations.

3.7 FLIGHT SERVICE

The flight service period in the final phase of the Demonstration Program Plan begins after the installation and test of the hybrid composites horizontal stabilizer (Paragraph 3.6.2) at an airline overhaul base during a regularly scheduled major maintenance period.

Upon satisfactory completion of the flight certification program, the airplane will be returned to the airline for regularly scheduled commercial airline service for a predetermined period of time with NASA concurrence.

Periodic inspection procedures for evaluating bond joint quality will be performed using contact pulse-echo or eddy-sonic test methods. The bond joint at the rear spar assembly and upper and lower skins can be externally checked by ultrasonic or eddy-sonic methods (See Figure 30).

The bond joint of the composite at the top of the stringer tees cannot be tested unless access is provided. Limited access is provided in the outer horizontal tail, box section, through two doors. Door #78 is 7 inch diameter in the upper surface and door #63 is 10 inch diameter in the lower surface. Both doors provide access to the elevator hinge pin. Access through the rear spar lightening holes is provided by lower surface doors, forward of the closure spar adjacent to the elevator actuators and hinges. Limited access can be provided through the lightening holes in the front spar in the area of the lower panel doors in the leading edge.

The external inspections of the upper and lower laminate to skin bonds along the stringers and rear spar can be easily performed. The laminate to stringer-cap bonds require access to the interior of the stabilizer necessitating removal of the leading edge and front spar ligntening hole access panels. Access to the aft internal section will require removal of lightening hole access panels in the rear spar.

Reference standards fabricated for the in-process inspection (Section 3.5) will need to be identified and saved for the periodic maintenance inspections.

After the airline flight service demonstration the airplane will be returned to the UAL maintenance base in San Francisco. The hybrid composite horizontal stabilizer will be removed, the all metal horizontal stabilizer reinstalled, and the necessary check out flight made, and the airplane returned to scheduled airline service.

Inspection, documentation, disassembly and test (if deemed necessary) and disposition of the hybrid composite horizontal stabilizer will be coordinated with the NASA, LRC.

3.8 COST-BENEFIT APPROACH

Data will be gathered on the basic elements of cost benefit evaluation which include engineering, development support, flight test operations, manufacturing, materials, tooling and quality control. The data will be evaluated by the analytical tools which have been modified and made available. See Appendix G.

3.9 STATEMENT OF WORK

The Douglas Aircraft Company, McDonnell Douglas Corporation, proposes to perform a Demonstration Program composed of five phases including the Phase I Study. An alternate plan composed of three phases including the Phase I Study is presented. The first plan, which is the recommended one, was presented at the final review held at the NASA, LRC, and accepted by the NASA as the preferred approach. The alternate plan was developed during Phase I.

3.9.1 The Five Phase Plan

The Feasibility Study (Phase I) as reported will be followed by four additional phases identified as Advanced Development Tests (Phase II), Detail Design and Analysis (Phase III), Component Fabrication and Test Plan (Phase IV), and Component Tests and Flight Service (Phase V). Phases II, III, IV, and V will be accomplished by five, seven, three and 13 integrated tasks, respectively.

<u>Phase II ADVANCED DEVELOPMENT TESTS</u>

Task II-A Review design for current state-of-the-art.

Task II-B Structural Design.

1. Evaluate several promising hybrid composite structural concepts.
2. Select the best structural concept.
3. Design sub-component specimen of composite/metal bonded joint for inboard end of outer skin panel.
4. Test drawings.

Task II-C Analysis

1. Establish criteria and loads.
2. Determine allowable strengths and constants.
3. Evaluate final bonded joint design of the outer skin panel.
4. Perform structural analysis (Report).

Task II-D Design Data Testing

1. Upgrade literature survey.
2. Selection of materials.
3. Prepare test plan.

Task II-D Design Data Testing (Continued)

4. Fabricate coupon specimens and full-scale subcomponent specimens.

5. Conduct tests.

6. Reduce test data (Report).

Task II-E Reporting and Management

1. Program Management.

2. Reporting.

<u>Phase III DETAIL DESIGN AND ANALYSIS</u>

Task III-A Structural Design

1. Layouts and production drawings-outer panel assemblies.

2. Layouts and production drawings-constant section.

3. Calculate weight reduction for the horizontal stabilizer.

4. Calculate net weight and balance change for the DC-8 airplane.

5. Test drawings.

Task III-B Analysis

1. Evaluate final design.

2. Perform structural analysis (Report).

3. Perform aeroelastic and flutter analysis (Report).

4. Formalize strength test requirements.

5. Improve analysis methods and procedures.

Task III-C Subcomponent Testing

1. Prepare test plan.

2. Fabricate test specimens.

3. Conduct subcomponent static and fatigue tests.

Task III-D Specification Preparation

1. Prepare specifications on methods and tolerances.

2. Prepare quality control and subcomponent test specs.

Task III-E Reliability and Quality Assurance Provisions and NDT

1. Implement reliability program provisions.

2. Implement quality program provisions.

3. Select best NDT methods.

4. Inspect components and assemblies using NDT.

Task III-F Operations Analysis

1. Collect, evaluate, and maintain Phase III cost data.

2. Cost data report.

Task III-G Reporting and Management

1. Program Management.

2. Coordinate with Airlines, FAA, and NASA.

3. Reporting.

Phase IV COMPONENT FABRICATION AND TEST PLAN

Task IV-A Fabrication and Assembly (Three horizontal stabilizers)

1. Establish fabrication and processing techniques.

2. Design and fabricate tooling.

3. Manufacturing and assembly sequence.

4. Fabrication and assembly of three horizontal stabilizers.

5. Flight flutter instrumentation on one flight test horizontal stabilizer.

Task IV-B Component Test Plan

1. Prepare test plan.

2. Obtain FAA and NASA approval of test plan.

3. Complete test drawings.

Task IV-C Reporting and management

1. Program management.

2. Reporting.

Phase V COMPONENT TESTS AND FLIGHT SERVICE

Task V-A Static Tests

Conduct static proof tests (100% limit load) on the three horizontal stabilizers (two flight and one ground test).

Task V-B Fatigue Test

Conduct fatigue test on ground test horizontal stabilizer.

Task V-C Ultimate Strength Test

Conduct ultimate strength test on ground test stabilizer after completion of fatigue test.

Task V-D Ship two horizontal stabilizers to the UAL Maintenance Base in San Francisco.

Task V-E Horizontal Stabilizer Installation

1. Remove horizontal stabilizers and elevators from two DC-8 airplanes.

2. Install new hybrid stabilizers on the DC-8 airplanes.

3. Install elevators and rig for flight.

Task V-F Ground Certification Program

1. Instrument horizontal stabilizer.

2. Conduct ground vibration tests.

Task V-G Flight Certification Program

1. Establish flight service plan.

2. Fabricate, assemble, and check out (Long Beach) of the instrumentation package.

3. Install and check out instrumentation package in airplane (San Francisco).

4. Conduct flight demonstration.

5. Data reduction and analysis.

Task V-H Obtain FAA type certificate

Task V-I Airline Flight Service

1. Prepare periodic inspection procedures for test surface.

2. Document and monitor data from airplane log and periodic inspections.

Task V-J Airplane Refurbishment

1. Remove test surfaces (horizontal stabilizer)

2. Install original stabilizers and elevators.

3. Acceptance flight.

4. Disposition of test parts.

Task V-K Operations Analysis

1. Conduct cost analyses based on cost data from Phases III, IV and V.

2. Cost benefit analysis report.

Task V-L Reporting and Management

1. Program management.

2. Coordinate with Airlines, FAA, and NASA.

3. Reporting.

3.9.2 The Three Phase Plan

The Feasibility Study (Phase I) as reported will be followed by two additional phases identified as Design, Analysis, Test, and Component Fabrication Program (Phase II) and Component Test and Flight Service (Phase III). Phase II will be accomplished by 11 integrated tasks and Phase III by nine integrated tasks.

<u>Phase II DESIGN, ANALYSIS, TEST, AND COMPONENT FABRICATION PROGRAM</u>

Task II-A Review design for current state-of-the-art

Task II-B Analysis

1. Establish criteria and loads

2. Determine allowable strengths and constants

3. Analyze configurations

Task II-B Analysis (Continued)

4. Evaluate final design
5. Perform structural analysis (Report)
6. Perform aeroelastic and flutter analysis (Report)
7. Formalize strength test requirements
8. Improve analysis methods and procedures

Task II-C Component Design

1. Layouts and production drawings - outer panel assemblies
2. Layouts and production drawings - constant section
3. Calculate weight reduction for the horizontal stabilizer.
4. Calculate net weight and balance change for the DC-8 airplane.

Task II-D Design Data Testing

1. Upgrade literature Survey
2. Selection of Materials
3. Prepare test plan
4. Fabricate and test specimens
5. Correlate and summarize test data

Task II-E Subcomponent Testing

1. Prepare test plan
2. Fabricate test specimens
3. Conduct subcomponent static and fatigue tests
4. Reduce test data (Report)

Task II-F Fabrication and Assembly

1. Establish fabrication and processing techniques
2. Design and fabricate tooling
3. Manufacturing and assembly sequence

Task II-F Fabrication and Assembly (Continued)

4. Fabrication and assembly

5. Instrument inaccessible areas with wire crack detectors and flight flutter instrumentation

Task II-G Component Testing

1. Prepare test plan

2. Conduct static test

3. Reduce test data (Report)

Task II-H Specification Preparation

1. Prepare specifications on methods and tolerances

2. Prepare quality control and subcomponent test specs

Task II-I Quality Control and NDT

1. Establish quality control procedure

2. Establish quality assurance procedures

3. Select best NDT methods

4. Inspect components and assemblies using NDT

Task II-J Operations Analysis

1. Collect, evaluate, and maintain Phase II cost data

2. Cost data report

Task II-K Reporting and Management

1. Program management

2. Coordinate with Airlines, FAA, and NASA

3. Monthly reports

4. Quarterly reports

5. Interim review

6. Final report

Phase III COMPONENT TEST AND FLIGHT PROGRAM

Task III-A Horizontal Stabilizer Shipped to UAL Maintenance Base in San Francisco

Task III-B Horizontal Stabilizer Installation

1. Remove horizontal stabilizer and elevators from DC-8 airplane
2. Install new hybrid stabilizer
3. Install elevator and rig for flight

Task III-C Ground certification Program

1. Instrument horizontal stabilizer
2. Conduct ground vibration tests with or without deflection tests

Task III-D Flight Certification Program

1. Establish flight demonstration plan
2. Fabricate, assemble, and check out (Long Beach) of the instrumentation package
3. Install and check out instrumentation package in airplane (San Francisco)
4. Conduct flight demonstration
5. Data reduction, analysis, and report

Task III-E Obtain FAA type certificate

Task III-F Airline Service Program

1. Prepare periodic inspection procedures for test surfaces
2. Document and monitor data from airplane log and periodic inspections
3. Flight program report

Task III-G Airplane Refurbishment

1. Remove test horizontal surfaces
2. Install original stabilizer and elevator
3. Acceptance flight
4. Disposition of test parts

Task III-H Operations Analysis

1. Conduct cost analyses based on both Phase II and III cost data
2. Cost benefit analysis report

Task III-I Reporting and Management

1. Program management
2. Coordinate with Airline, FAA, and NASA
3. Monthly reports
4. Quarterly reports
5. Interim reports
6. Final reports

3.10 PROGRAM SCHEDULE

The performance schedule that is proposed for accomplishing the four additional phases of the five phase Recommended Program is presented (Figure 31).

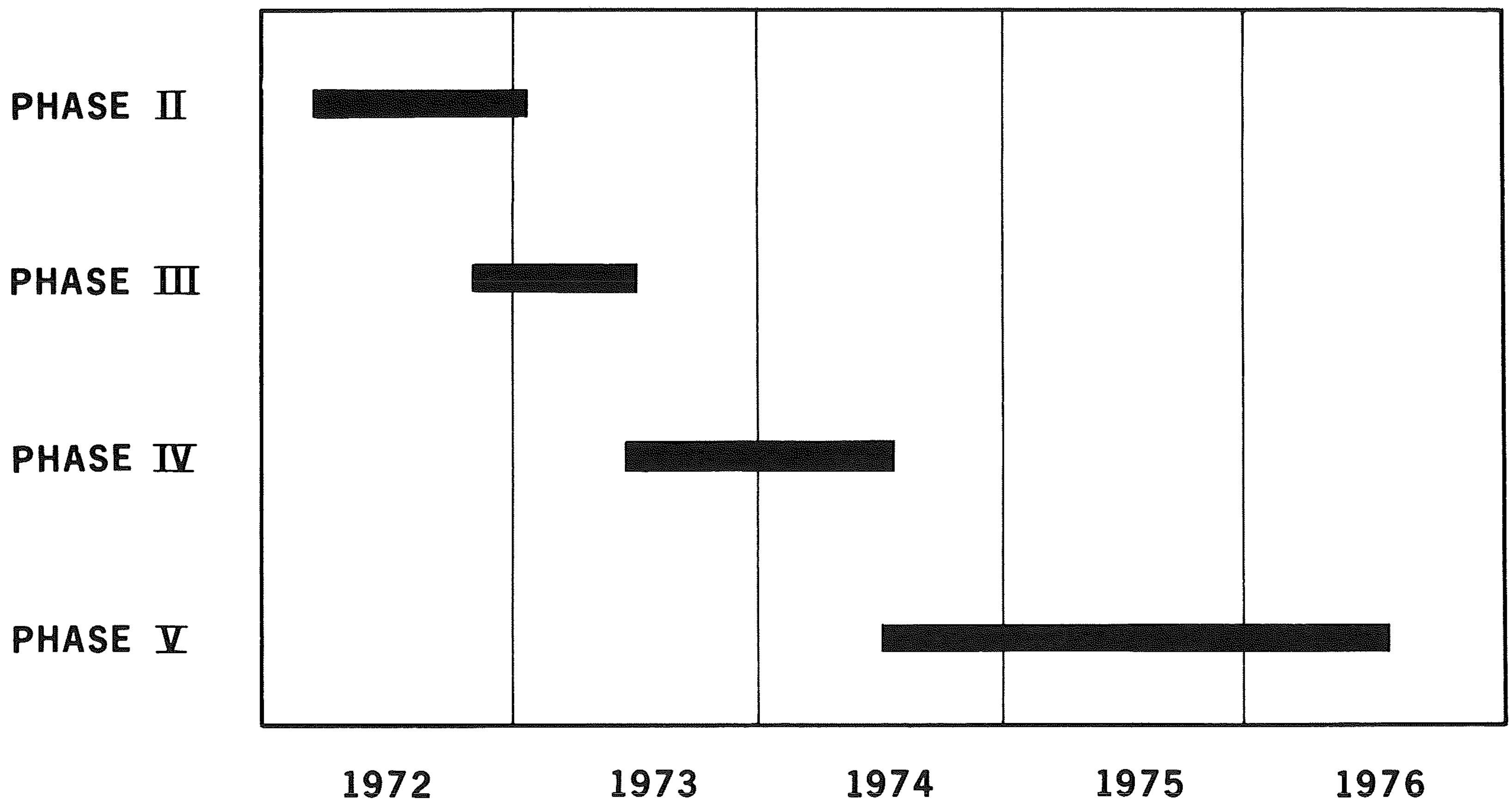


FIGURE 31. PROGRAM SCHEDULE

APPENDIX A

COMPONENT EVALUATION AND SELECTION

The DC-8 airplane was initially chosen as the vehicle for consideration of hybrid composite components. Shortly after Phase I activities began and some evaluation of the components on the DC-8 had been made, it was determined that both the DC-9 and DC-10 airplanes would merit some consideration in terms of a potential application for hybrid composites.

The DC-9 horizontal stabilizer looked especially promising for several reasons: (1) the horizontal stabilizer is a one piece assembly (DC-8 part consists of three major units), (2) all bonding could be accomplished in-house whereas the size of the DC-8 parts would require that autoclave facilities outside Douglas be used, and (3) the down-time associated with removal and installation of the stabilizer is approximately one-half the cost of the DC-8 stabilizer. The over-riding single factor which eliminated the DC-9 from further consideration was the impact a change to the horizontal stabilizer would have on the vertical stabilizer. A preliminary analysis indicated that substantial analyses and possible tests would be required on the vertical stabilizer to ensure that the structure was still flightworthy. Consequently, the choice of the horizontal stabilier would require analysis and tests for not only this component but also the vertical stabilizer with associated increases to both program schedules and costs. The DC-9 airplane was eliminated as a potential vehicle for hybrid composites application for the reasons mentioned.

The DC-10 airplane was also given some consideration. However, due to the larger size of parts, higher costs entailed in fabrication, and the lack of background comparison data, the DC-10 was eliminated as a serious possibility for hybrid composites application for this particular program.

Five candidate components of primary structure on the DC-8 commercial jet liner were proposed for detailed consideration to determine which was best suited for design, analysis, and construction as a hybrid composite. The five components, shown in Figure A-1, were the wing outboard box, the inboard and outboard pylons, the horizontal stabilizer and the vertical stabilizer.

1. SELECTION CRITERIA

Criteria were established as a basic guideline for assessing the relative merits of each of the five components. The criteria are divided into six major areas.

- o Primary structure of significant size.
- o Reasonable weight saving afforded by application of composite elements.

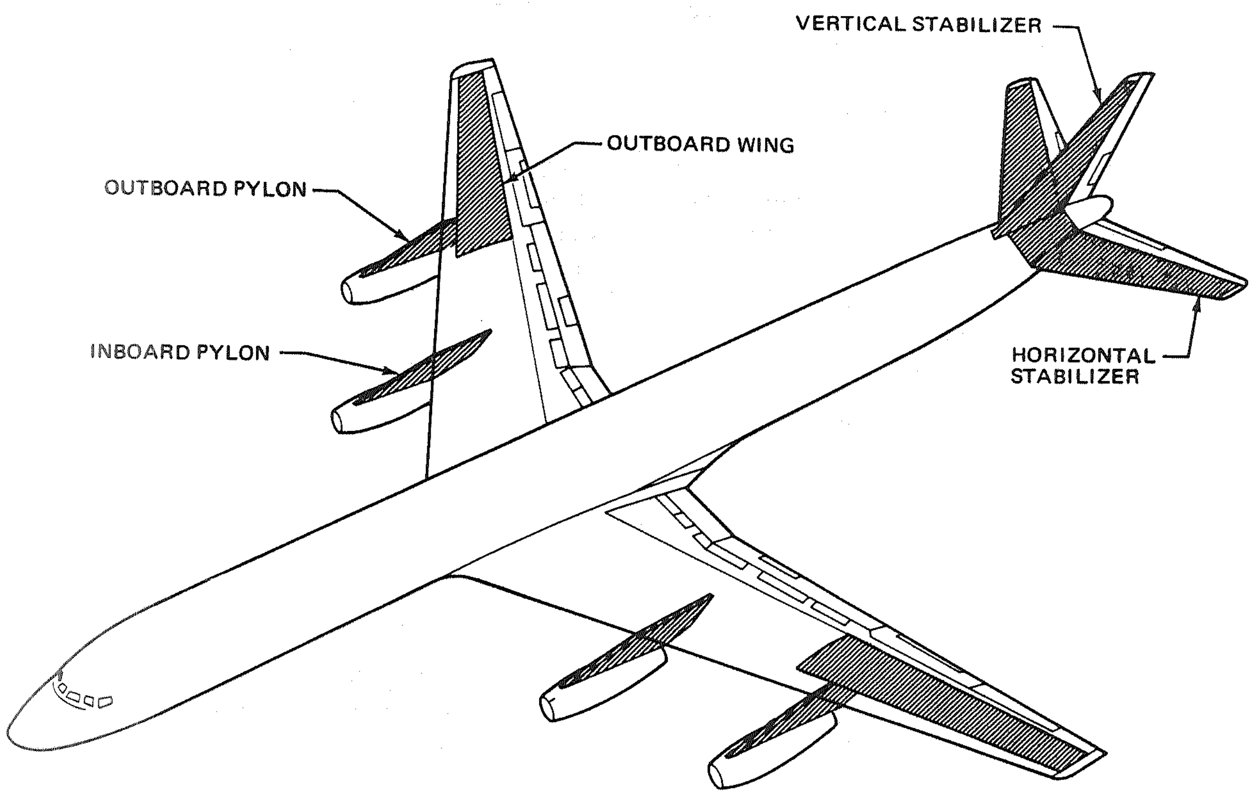


FIGURE A-1. CANDIDATE COMPONENTS

- o Must incorporate failsafe design.
- o Interchangeable or replaceable assembly.
- o Assembly not unduly complicated by application of composites.
- o Reasonable program cost.

2. CANDIDATE COMPONENTS

Horizontal Stabilizer

The basic load carrying portion of the stabilizer is made of a two spar structural box which passes uninterrupted through the fuselage to eliminate any fatigue sensitive discontinuities. This structural box is attached to the airplane fuselage by two pivot points at the rear spar and two screw jacks at the front spar which provide support for the entire horizontal stabilizer and also permits trim adjustment. Each of the upper and lower skins is comprised of three integrally stiffened skin planks which are fastened along a spanwise joint to provide failsafe capability. The rib assemblies are spaced approximately 24 inches apart.

Vertical Stabilizer

The vertical stabilizer is a three spar structure which is manufactured as an integral part of the fuselage aft section. The skin is of double sheet construction with the inner sheet beaded chordwise to add rigidity and to provide a double load path for the torque shear. The ribs are spaced approximately 30 inches apart. The upper portion of the vertical stabilizer constitutes the HF antenna and is electrically isolated from the rest of the stabilizer by a fiberglass laminate isolation band.

Outboard Wing

The wing structure is of failsafe design with three spars so that if one spar should fail the remaining two spars will provide adequate strength. The stringers are separate rather than integral with the skin. The spar caps and stringers are spliced near the skin splice which is located at about midpoint on the semi-span, slightly inboard of the outer engine pylons. Skin and spar caps continue through the side of the fuselage so that the main load carrying splice is on the centerline, inside the airplane. The entire space between front and rear spars from tip to tip is utilized as an integral fuel tank.

Inboard Pylon

The inboard engine pylon internal structure is an assembly of five spars, a keel structure and ribs. A horizontal titanium firewall extends the full length of the lower end of the pylon. The engines are supported on the bottom of the pylon at two points on the engine mount support structure which ties into the spars through bulkheads. The pylon is attached to the wing through the front spar to the wing bulkheads and to chordwise attach fittings on the wing skins.

Outboard Pylon

The external appearance of the outboard pylon is similar to the inboard pylon; however, in order for the outboard pylon to function complimentary to the other aeroelastic characteristics of the airplane, this pylon is more flexible than the inboard pylon. A flexible joint is designed into the structure so that the forward part of the pylon is permitted to deflect independently of the keel structure in the aft part of the pylon.

The pylon is basically a two spar structure with bulkheads, skins, and a titanium firewall. The engine support and wing attach structures are similar to the inboard pylon.

3. COMPONENT SELECTION

In selecting the final component for the investigation of the feasibility of hybrid composite application, advantages and disadvantages of the candidate component were considered, and they are listed in Table A-I.

The wing was eliminated principally because of three factors: the long aircraft down time was prohibitively expensive, there was doubt that the structure would fit properly because of residual stresses, particularly since two disassemblies and reassemblies were involved and the weight savings would be less than on the other candidates because of stringent fatigue requirements.

The pylons were eliminated because of a much more severe thermal environment, up to 250°F, which would have required the use of technology beyond the established in-house capabilities.

The vertical and horizontal stabilizers have similar structures and correspondingly alike advantages and disadvantages as a hybrid, but the vertical stabilizer is structurally integral with the rear fuselage whereas the horizontal stabilizer is a discrete structure attached by the mechanical actuating system. The horizontal stabilizer, therefore, afforded the advantages of a structure not critical in fatigue, a separate component which could be fabricated completely and tested independent of the aircraft, the shortest down time during changeover from a metal to the hybrid stabilizer and vice versa, a design within the in-house capabilities of fabrication, and the greatest potential weight savings.

On the basis of a far more comprehensive and detailed investigation than the outline above, it was decided to recommend to the NASA, LRC, that the horizontal stabilizer be the candidate component selected. NASA concurred with the Douglas recommendation and the horizontal stabilizer was selected as the component for further study.

TABLE A-I

SUMMARY OF ADVANTAGES AND DISADVANTAGES FOR EACH COMPONENT

COMPONENT	ADVANTAGES	DISADVANTAGES
Horizontal Stabilizer	o Straightforward design o Simple joint to fuselage o Short down time for removal and installation o One structural configuration	o Because of its simplicity, it leaves unresolved problems peculiar to other structures on the aircraft
Vertical Stabilizer	o Straightforward design o One structural configuration	o Integral part of fuselage o New installation jigs required o Some redesign and testing for antenna systems
Outboard Wing	o Largest structure	o Lightning critical o Fuel environment o Fuel sealing problems o Residual stresses and warpage during removal and installation o Complicated shear joint o Most testing required o Many structural configurations o Model testing required o Most analysis required o Costly design o Many service lines
Pylons	o Short down time for removal and installation	o Costly design o Elevated temperature o Lightning critical o Many structural configurations o Severe material problems

APPENDIX B

WEIGHT SAVINGS ANALYSIS

An evaluation was made to assess the feasibility of incorporating the hybrid composites concept on the various major assemblies and detail parts for the entire horizontal stabilizer structure. Some of the assemblies and parts such as the leading edge assemblies and trailing edge boxes were designed by minimum gage requirements rather than by strength and/or stiffness requirements. The operating bulkhead which attaches the horizontal stabilizer to the fuselage and is located between and joins the constant section center box and outboard structural box sections together is essentially a heavy shear plate with fittings and did not lend itself to hybrid composites application. In addition, most of the ribs and the shear webs for the spar assemblies have large lightening holes and are either minimum gage or close to minimum gage. The outboard front spar cap consists of three flanges (two horizontal and one vertical). The flanges attach the leading edge, integrally stiffened skin planks, and front spar shear web. The attachments made it difficult to avoid clearance problems with mechanical fasteners and to locate boron-epoxy on the cap. A summary of the weight savings analysis is shown in Table B-I.

TABLE B-I

SUMMARY OF WEIGHTS

ITEM	WEIGHT	ALL METAL (lb)	HYBRID COMPOSITE (lb)	REDUCTION (lb)	REDUCTION (%)
CONSTANT SECTION CENTER BOX	SKIN PANELS	109.7	93.4	16.3	14.9
	FRONT SPAR: ASSEMBLIES	35.1	32.1	3.0	8.6
	CAPS (PART OF ASSY)	[15.8]	[12.8]	[3.0]	19.0
	REAR SPAR: ASSEMBLIES	48.8	42.8	6.0	12.3
	CAPS (PART OF ASSY)	[31.9]	[25.9]	[6.0]	18.8
	OTHER STRUCTURE	12.0	12.0	—	—
OUTBOARD STRUCTURAL BOX	SKIN PANELS	454.2	400.3	53.9	11.8
	REAR SPAR: ASSEMBLIES	209.5	190.3	19.2	9.2
	CAPS (PART OF ASSY)	[119.7]	[100.5]	[19.2]	16.0
	OTHER STRUCTURE	332.7	332.7	—	—
HORIZONTAL STABILIZER STRUCTURAL BOX		1202.0	1103.6	98.4	8.2

NOTE: [ ] DENOTES WEIGHTS WHICH ARE A PART OF SPAR ASSEMBLY WEIGHTS

APPENDIX C

MATERIALS ANALYSIS AND COSTS

I. STATUS OF MATERIALS TECHNOLOGY

A. Prepreg Materials

A brief survey was made of existing high modulus reinforced prepregs to establish potential candidate composite materials for use during the detail design, analysis, test, and component fabrication phase. Table C-I shows typical properties of the unimpregnated reinforcements and Table C-II displays typical property values exhibited by unidirectional composites made from these reinforcements with an epoxy resin matrix. Note that three boron/epoxy products are listed in Table C-II. The first of these represents the product that has the greatest commercial exposure and for which large amounts of data are available. The second boron prepreg is listed because of its unique low temperature curing capability (250°F) even though only limited data is available. The third boron/epoxy uses the 0.0056 inch diameter fiber instead of 0.004 inch. The primary advantages of the third system are: (1) the projected lower raw material cost and (2) some reduction in lay-up time.

B. Adhesives

As in the case of the prepregs, a survey was made of available adhesive systems pertinent to this program. Table C-III is a summary of the data collected.

While there are many chemical formulations for adhesives, from elasto-mechanical considerations they may be classified in two types: (1) the largely unmodified polymers, such as the epoxies and polyimides, which are relatively brittle and fracture at small shear strains; and (2) the plasticized polymers, such as the epoxy nylons, epoxy nitriles and vinyl phenolics, which are quite ductile and do not fracture until a considerable shear strain has been developed. Of the two the ductile adhesives permit stronger joints while the brittle adhesives induce lower thermal stresses.

Various test results on bonded hybrid joints indicate what can, and cannot, be accomplished. Oken and June (Boeing) tried modifying the cure cycle for 250°F curing ductile adhesives by decreasing the temperature and increasing the time. At Douglas, hybrid bonded joints using a brittle high-temperature curing adhesive frequently failed to survive cooling down to room temperature. The properties of this adhesive are close to those of the conventional resin matrices. Such tests, therefore,

TABLE C-I

NOMINAL REINFORCEMENT PROPERTIES

REINFORCEMENT / PROPERTY	BORON	ENGLISH GRAPHITE		UNION CARBIDE THORNEL		CELANESE	S-GLASS
	4 MIL	TYPE I	TYPE 2	50S	75S		
TENSILE STRENGTH ($PSI \times 10^{-3}$)	450	300	400	285	345	400	600
SPECIFIC STRENGTH ($INCH \times 10^{-6}$)	4.7	4.3	6.3	4.7	5.0	5.6	6.5
MODULUS ($PSI \times 10^{-6}$)	58	50	38	50	79	> 70	12
SPECIFIC MODULUS ($INCH \times 10^{-6}$)	610	714	600	820	1295	> 990	130
DENSITY (GM/CC) ($LB/IN.^3$)	2.63 0.095	1.95 0.070	1.74 0.063	1.68 0.061	1.82 0.069	1.95 0.071	2.49 0.092

TABLE C-II

REPRESENTATIVE PROPERTIES - UNIDIRECTIONAL COMPOSITES

Composite / Property	Boron/Epoxy			Boron/ Polyimide	Boron/Metal		Typical Graphite/Epoxy					Glass/Epoxy Scotchply
							Pan Fibers			Rayon Fibers		
	5505 (4mil) (Narmco)	SP-295 250° F (4mil) (3M)	SP-272 (6mil) (3M)		6061 Al	AZ92 Mag	Type I	Type II	Hi-Mod	50S	75S	1002 (S-Glass) (3M)
Reinf (% Vol)	50	-	50	50	50	65	58	60	71	60	57	64
Ult. Tensile Strength (psi x 10^{-3})	200	204	210	160	165	300	150	170	115	148	214	200
Tensile Mod. (psi x 10^{-6})	32	30	30	33	34	42	30	20	45	29	44	8.4
Comp. Str. (psi x 10^{-3})	300+	-	450	-	185	-	70	150	114	100	97	120
Horiz. Shear Strength (psi x 10^{-3})	14.5	14.5	13.0	12	16	21.4	12	16	9	8	6	11.2
Density (Lb/In^3)	0.070	0.069	-	0.072	0.10	0.080	0.059	0.053	0.061	0.052	0.057	0.072

TABLE C-III

SUMMARY OF CANDIDATE ADHESIVES FOR DC-8 HORIZONTAL STABILIZER

ITEM	IDENTIFICATION	MANUFACTURER	GENERIC CLASS	DUCTILITY	LAP SHEAR STRENGTH (PSI) $-67^{o}F$	RT	$+180^{o}F$	T-PEEL STRENGTH (PIW) $-67^{o}F$	RT	$+180^{o}F$	CURE CYCLE	PRESENT USAGE (DACO)	COMMENTS
1	EC-2214 Hi-Flex	3M Company	One-part liquid epoxy	?	3000	4500	2300	4	9	14	3 hrs at $200^{o}F$ 22 min at $250^{o}F$	None	No stress-strain data
2	EA-951	HYSOL	Epoxy nylon film	Very good	6515	6320	4270	38	85	40	1 hr at $350^{o}F$	DC-9 honeycomb & metal-to-metal bonded assys	Stress-strain curves available for room temp. Has shown good resistance to environmental exposure
3	FM-1000	BLOOMINGDALE	Epoxy-nylon film	Very good	7400	7090	3670	-	60	-	60 Min at 350 F	DC-9 honeycomb & metal-to bonded assys	Stress-strain curves available at room temp. Test have shown lower resistance to humidity than Epon 951
4	AF-32	3M Company	Modified epoxy film	Good	5086	4085	2286	10	60	25	1 hr at $250^{o}F$ 5 min at 350	Metal-to-metal & metal to fiberglass bonds	No stress-strain data. Requires a primer for proper adhesion.
5	AF-44	3M Company	Epoxy-nylon film	Very good	6500	6500	3000	32	73	34	60 min at 265 F	None	Stress-strain curves available at room temp (new adhesive)
6	AF-126	3M Company	Epoxy-nitrile film	Good	4850	4630	3850	24	36	22	1 hr at 250 F	None	Stress-strain curves available
7	FM-123-5	BLOOMINGDALE	Epoxy nitrile film	Good	5815	5020	3600	-	29	-	90 min 200 F 15 min 250 F	Metal-to-metal bonded assys	Stress-strain curves available for room temp. Use BR127 corrosion resistant primer
8	EA-9602	HYSOL	Modified epoxy	Good	5080	6500	4450	19	38	43	1 hr at 250 F 2 1/2 hrs at 200 F	None	No stress-strain curves available

indicate the need for a layer of suitable adhesive between the metal and the composite. Douglas experience in bonding hybrid structures has tended to encourage the use of 250°F curing ductile adhesives. An associated joint strength reduction of from 20 to 40 percent attributable to effects of thermal imbalance is tolerated because the remaining strength was found to exceed the total strength of room temperature curing adhesives.

A further factor favoring the use of such ductile adhesives is that doing so enables a far greater redistribution of load around a stress concentration than is possible with a brittle adhesive. This improves the fatigue behavior of hybrid composites. Still greater ductility and joint strength can be obtained by using a ductile adhesive which cures at 350°F, but the increase in residual stresses in the metal outweighs the benefits from the improved joint strength. The load-transfer joints at the ends of the composite reinforcement must be carefully designed as scarf or stepped-lap joints to minimize the strain concentrations in the adhesive. The abrupt discontinuity around a debond or a hole in the composite imposes severe loads on the adhesive.

II. MATERIAL COSTS

Graphite and boron prepreg current and future estimated material cost information has been obtained from 18 different material suppliers. A set of combination curves incorporating the data supplied by the many companies has been drawn for both boron and graphite prepregs. The variance in costs and future cost estimates are based on several assumptions: (a) quantity ordered, (b) production volume, (c) type of fibers, (d) package form of prepreg, (e) company confidence on breakthrough of processing techniques, and (f) individual company pricing policy. See Figure C-1. The sharp drop in boron pricing which occured during the early 1971 period is based on larger diameter ($\sim$6 mil) boron fibers. An additional drop in boron prices may be expected if the tungsten carrier should be replaced by glass or carbon type fibers.

III. MATERIALS SELECTION

A firm recommendation for composite and adhesive materials can not be made at this time because of the rapid changes and continual improvements in material capabilities occuring in advanced composites. Advantageous developments will very likely occur during the procurement lead time from Phase I to Phase II. However, the investigations performed to date in this program have narrowed the selection considerably. If selection were to be made at this time, the following materials would be chosen:

A. Composites

Due to better thermal compatibility, boron-epoxy is highly

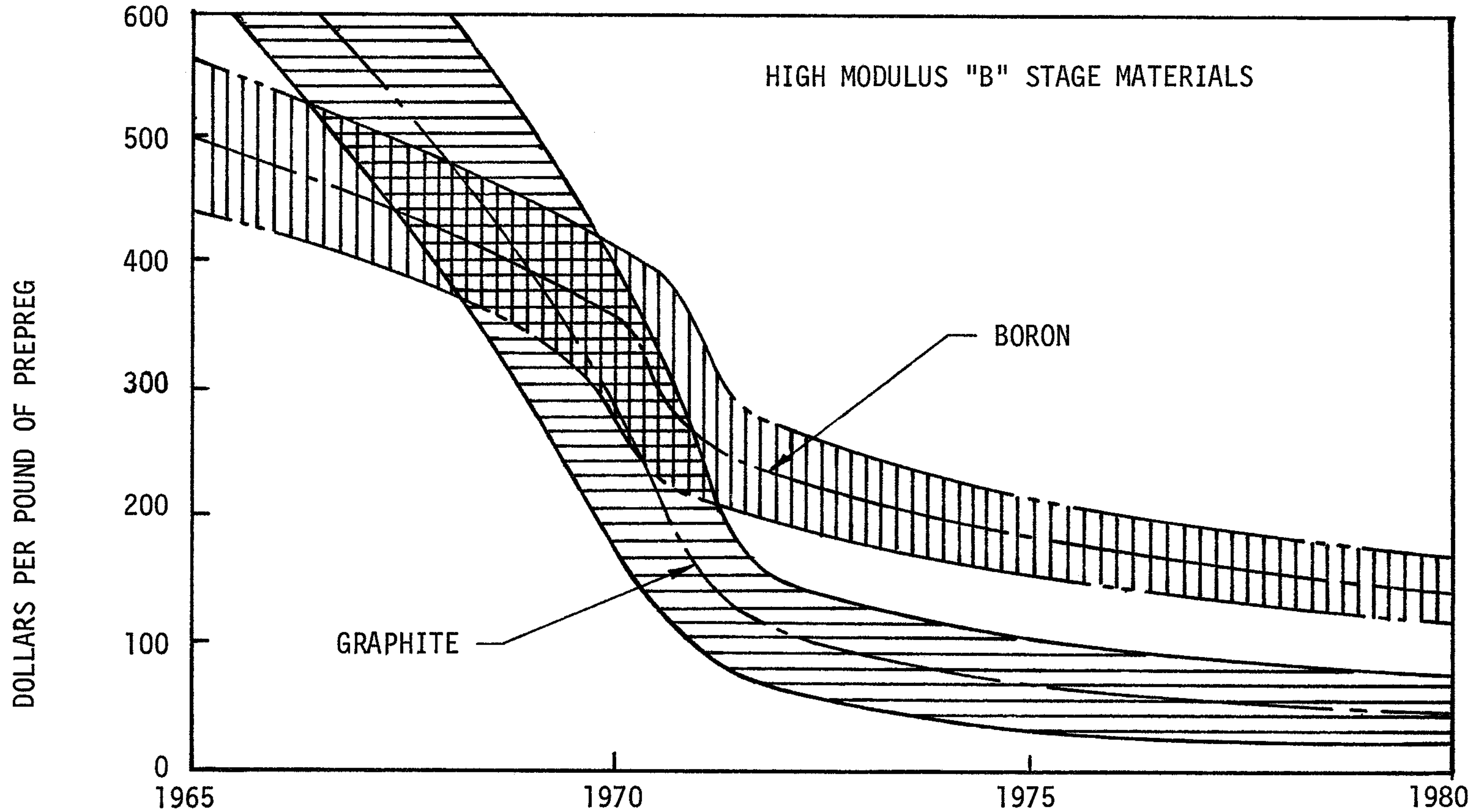


FIGURE C-1. PROJECTED COSTS OF BORON AND GRAPHITE PREPREGS preferred over graphite-epoxy. A combination not now readily available but with little risk would be the six or eight mil boron fiber (for reasons of cost) and the 3M-SP295 epoxy resin (for low temperature curing).

B. Adhesives

FM-123-5 nitrile modified epoxy from Bloomingdale is the leading candidate adhesive system because of extensive data and past experience. The material cures over a wide temperature/time range including temperatures as low as 180°F. A corrosion inhibitive primer, such as Bloomingdale BR-127 for the FM-123-5 system, must be used to reduce the strength degradation effects of moisture with time in the aluminum substrate.

The adhesives to be considered fall into two categories: (1) those 250°F curing ductile adhesives which could possibly be used alone, and (2) the pairs of adhesives (350°F curing ductile adhesive and 200°F curing brittle adhesive) which could be used in the bi-adhesive bonding system (to attain high strengths while alleviating thermal stresses). Room temperature curing adhesives would reduce the thermal stresses, but their shear strengths and environmental resistances are inadequate to the task. Room temperature curing acrylic adhesives offer considerable improvements over epoxies in these regards but they are considered too new to recommend for this program.

The 250°F curing adhesives for use alone are either epoxy-nylons or epoxy-nitriles. They all have generally similar stress-strain characteristics in shear and peel. The various adhesives are documented to different degrees and the Douglas Aircraft Company has had different levels of experience with each. The best documented is FM-123, while that with the greatest use at Douglas is AF-32. The AF-126 adhesive is of particular importance because of work by the 3M Company to modify it for use as both a matrix and adhesive in co-cure and bond applications.

The selection of adhesives for bonding together hybrid composites depends as much on the thermally-induced locked-in stresses as it does on the operating environment. The reason for this is that thermally induced stresses occur when materials of differing thermal expansion coefficients are joined at one temperature but must operate at another temperature. These locked-in stresses must be balanced by shear stresses induced in the adhesive if structural integrity is to be maintained. Such bond stresses alternately add to, and subtract from, the bond stresses due to external loads at the ends of each bond, as shown in Figure C-2. This reduces the capacity of the bond to react the externally applied loads. In the case of ductile adhesives, it is the shear strains rather than the stresses which limit the load capacity.

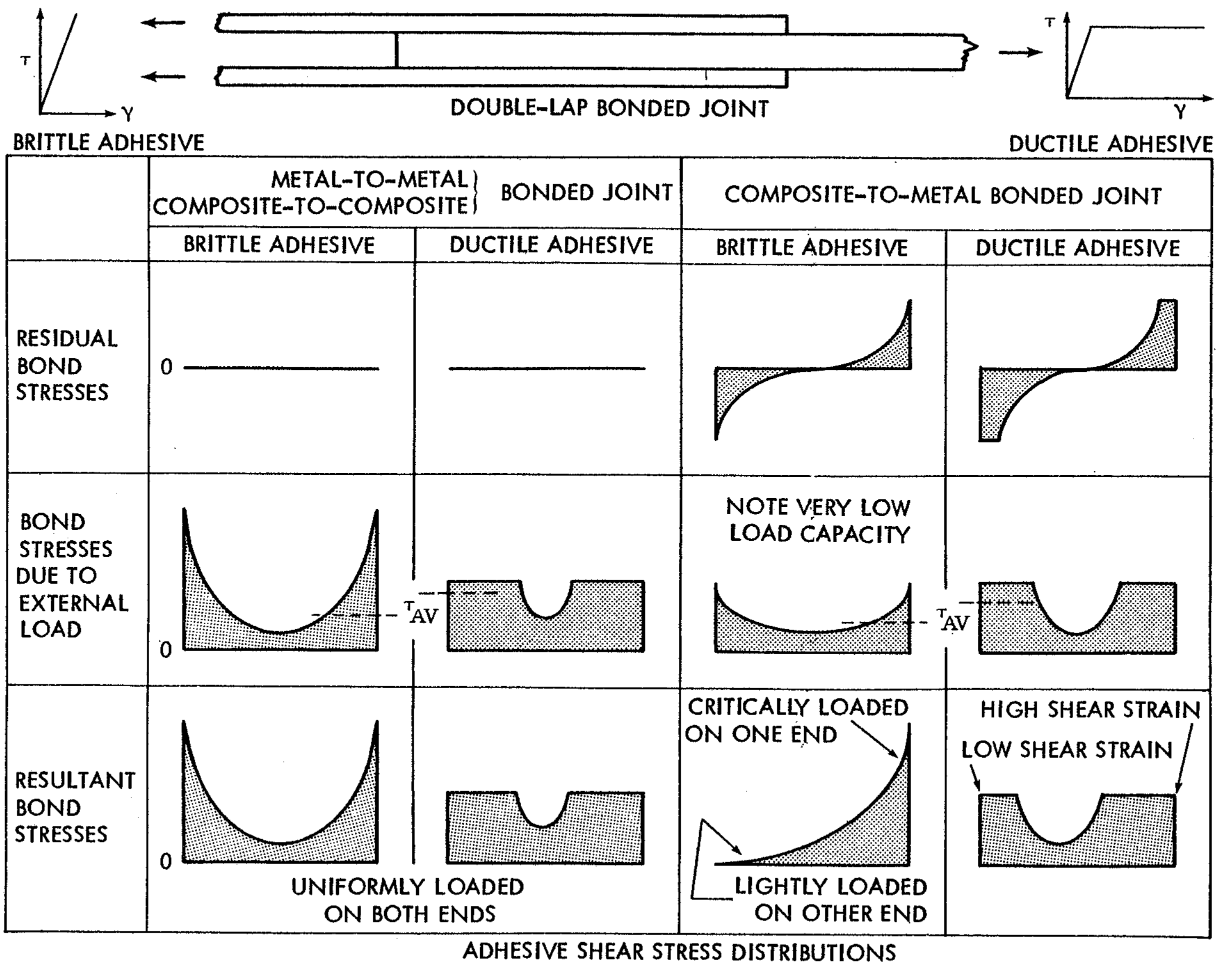


FIGURE C-2. ADHESIVE SHEAR STRESS DISTRIBUTIONS IN HYBRID JOINTS, COMPARING BEHAVIOR OF DUCTILE AND BRITTLE ADHESIVES

IV. RECOMMENDATIONS FOR ADDITIONAL DEVELOPMENT

Present state-of-the-art materials data and knowledge permit only a tentative selection of materials due to the unique requirements of this program. As a consequence, development work is recommended in the following areas:

A. The effects of galvanic action between the composite reinforcement and the aluminum. This is particularly significant for both boron and graphite composites (9).

B. Long term aging effects on properties at maximum service temperatures. Current information indicates a rather severe reduction of elevated temperature properties (>300°F) of boron and graphite epoxy composites after exposure to ambient conditions for 2 or 3 months. Whether there is a significant reduction at 180°F is not known.

C. Lower cost materials such as six or eight mil boron fiber prepreg. These materials should be capable of direct substitution in the program as soon as sufficient data is available to characterize them.

D. Generation of fatigue data on hybrid structures when subjected to service environments such as humidity, low and high temperatures, fuels, etc.

E. Ideally the chosen adhesive should: (1) have high stress resistance with high strain (stress-strain data at -67°F, RT, and +160°F is not available for all candidate adhesives), (2) cure at low temperatures (eg. halfway between -65°F and +180°F) (3) resist all typical environmental effects and (4) be compatible with the composite matrix curing mechanism.

F. Resin compatiblity. A brief confirmation study should be made to verify the chemical, physical and cure cycle compatibility of the matrix resin and the adhesive.

APPENDIX D

BI-ADHESIVE BONDING CONCEPT

A Douglas disclosure item aimed at improving the load transfer in bonded joints between dissimilar materials is shown in Figure D-1. This process called bi-adhesive bonding is used to alleviate the high thermal stresses normally associated with the use of high-strength adhesives, all of which cure at elevated temperature in the bonding together of dissimilar materials. One advantage is a reduction in the residual tensile stresses normally induced in the metal as the hybrid cools down from the cure temperature to the operating temperature and, in consequence, an improved fatigue life. Another advantage is that a stronger joint can be achieved and the filamentary composite is protected against interlaminar shear failure. The ductile adhesive is first bonded to the composite to provide a high joint strength and this adhesive is then bonded to the metal with an adhesive curing at a much lower temperature. The latter will probably be brittle because it must be capable of developing a higher peak shear stress than does the ductile adhesive (when each is used alone), but is protected from failure by the shear cutoff enforced by the ductile adhesive. If a single low temperature curing adhesive is used alone, the joint strength transferrable would be limited greatly by the lack of ductility of the adhesive.

The bi-adhesive system uses either EA-951 or FM-1000 for the ductile adhesive (both have been used extensively and interchangeably at Douglas) and EC-2214 Hi-Flex for the low-temperature curing adhesive layer.

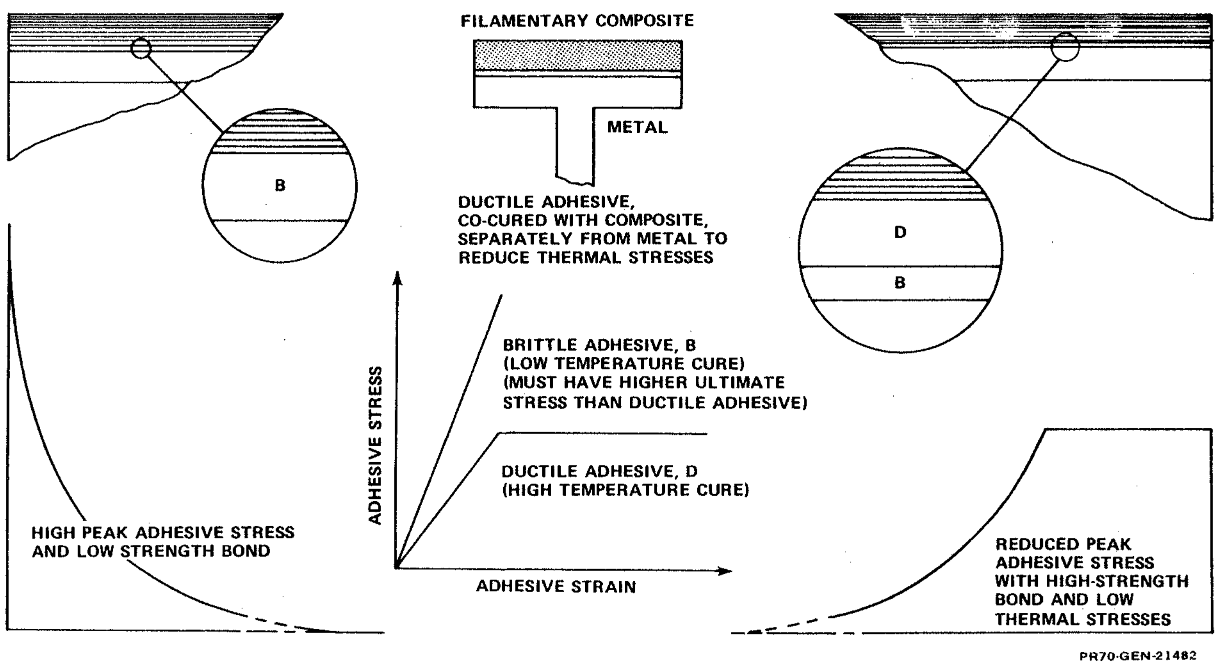


FIGURE D-1. REDUCTION OF PEAK ADHESIVE STRESS BY BI-ADHESIVE BONDING FOR HYBRID COMPOSITE JOINTS.

APPENDIX E

PROCESS CONTROL PROCEDURES AND NDT

I. QUALITY CONTROL

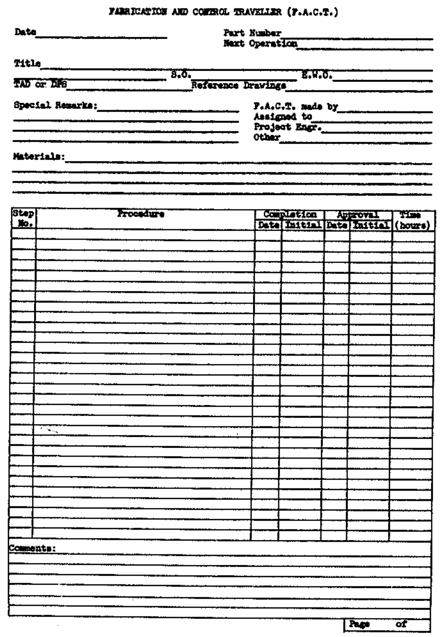
A review of the quality control procedures which have been developed over the many years of Douglas experience for the fabrication and assembly of metal and composite structures indicate that adequate procedures exist for producing high quality parts. First the prepreg materials which are received for use will be checked for prepreg resin content, prepreg volatile content, gel time, flexural strength and modulus, and short beam shear. Second, adhesives will be checked for tensile lap shear strength.

Fabrication and Control Traveller (FACT) sheets have been in use for a number of years to control the fabrication of parts by a step-by-step callout of each operation to be performed. The FACT sheet is prepared for each significant phase of test component and end-item manufacture. An example of a FACT sheet is shown in Figure E-1.

The quality control requirements will be detailed for each step of the operation including appropriate instructions on each FACT sheet as well as any necessary upgrading of material specifications.

II. NDT METHODS

A literature search was conducted on available nondestructive test (NDT) methods and an assessment made of the various potentially applicable methods for detecting debonds, fractures within the laminate, and cracks. See Table E-I for a summary of NDT methods to be used for the determination of the various kinds of defects in boron-epoxy and other composite materials (10).



FABRICATION AND CONTROL TRAVELLER (F.A.C.T.)

Date_____ Part Number_____
Next Operation_____

Title_____
_____ S.O._____ E.W.O._____
TAD or DPS_____ Reference Drawings_____

Special Remarks:_____ F.A.C.T. made by_____
Assigned to_____
Project Engr._____
Other_____

Materials:_____

Step No.	Procedure	Completion		Approval		Time (hours)
		Date	Initial	Date	Initial	

Comments:

Page ___ of ___

FIGURE E-1. EXAMPLE OF "FABRICATION AND CONTROL TRAVELLER" FORM

TABLE E-1

APPLICATIONS AND LIMITATIONS OF NONDESTRUCTIVE TEST METHODS TO COMPOSITE MATERIALS

DEFECT	FIBER GLASS	BORON FIBER	GRAPHITE FIBER
UNBONDS	SONIC	SONIC	SONIC
	ULTRASONIC	ULTRASONIC	ULTRASONIC
	THERMAL (THIN SECTIONS ONLY)	THERMAL (THIN SECTIONS ONLY)	
		EDDY-SONIC	EDDY-SONIC
DELAMINATIONS	SONIC	SONIC	SONIC
	ULTRASONIC	ULTRASONIC	ULTRASONIC
		THERMAL (THIN SECTIONS ONLY)	
		EDDY-SONIC	EDDY-SONIC
FIBER-ORIENTATION (TAPE)	BACK-LIGHTING (MAGNIFIED)	BACK-LIGHTING (MAGNIFIED)	
	X-RAY (LOW KV)	X-RAY (LOW KV)	X-RAY (LOW KV)
FIBER-ORIENTATION (LAMINATE)	X-RAY	X-RAY	
	MICROSCOPE (EDGE)	MICROSCOPE (EDGE)	MICROSCOPE (EDGE)
INCLUSIONS	X-RAY	X-RAY	X-RAY
	ULTRASONIC	ULTRASONIC	ULTRASONIC
CRUSHED HONEYCOMB CORE	X-RAY	X-RAY	X-RAY
	ULTRASONIC (THROUGH-TRANSMISSION)	ULTRASONIC (THROUGH-TRANSMISSION)	ULTRASONIC (THROUGH-TRANSMISSION)
		EDDY-SONIC	EDDY-SONIC
RESIN-RICH AND RESIN-STARVED AREAS	X-RAY	X-RAY	X-RAY
	ULTRASONIC	ULTRASONIC	ULTRASONIC
MICROMECHANIC STUDIES	ACOUSTIC EMISSION	ACOUSTIC EMISSION	ACOUSTIC EMISSION
THICKNESS GAGING	MICROMETER	MICROMETER	MICROMETER
	ULTRASONIC	ULTRASONIC	ULTRASONIC
POROSITY AND CRACKS (INTERNAL)	ULTRASONIC	ULTRASONIC	ULTRASONIC
	X-RAY	X-RAY (THIN SECTIONS ONLY)	X-RAY (THIN SECTIONS ONLY)
POROSITY AND CRACKS (EXTERNAL)	PENETRANT	PENETRANT	PENETRANT

APPENDIX F

HORIZONTAL STABILIZER REMOVAL AND INSTALLATION

The major steps and operations required for the removal and installation of the DC-8 horizontal stabilizer are outlined below.

I. REMOVAL SEQUENCE

A. <u>Elevator Removal</u>

Install hoisting sling and neutralize weight.

Remove 13 access doors and fairings from hinge cutouts; Open circuit breakers.

Disconnect cranks on dampers, trim tabs, stop bolts.

Remove torque tube for tab drive.

Remove bolts attaching elevator to inboard hinge fitting.

Remove elevator assembly and store.

Left and Right Assemblies are identical.

B. <u>Stabilizer Removal</u>

Remove fairing and install support on panel not being removed.

Install hoisting sling and take up slack (See Figure F-1).

Remove brush baffle assembly, connector to de-icing duct, and 90 bolts to center section.

Move panel out six to eight inches and cut electrical wires, tape and tag.

Left and Right Assemblies are identical.

C. <u>Center Section Removal</u>

Place 3 jacks under constant section center box.

Remove hinge shaft (preload stressed shaft requires special tools).

Disconnect longitudinal trim drive mechanism anti-twist bar.

Remove operating bulkhead from center section.

Remove stabilizer electrical wiring from center section conduit.

Lift out constant section (see Figures F-2 and F-3).

II INSTALLATION SEQUENCE

A. Center Section Installation

Check jack locations and place center section on jacks. (See Figures F-2 and F-3)

Install operating bulkheads on center, adjusting jacks to align.

Install hinge shaft, pilot, spacers and laminate washers.

Tighten hinge shaft outboard nut to 4600 Inch-Lb and safety.

Tighten inboard shaft nut to 550 inch-lb.

Remove jacks and install anti-twist bar.

Install travel limit stop link stops and bracket.

Replace wiring within conduit.

B. Stabilizer Installation

Sling stabilizer into position for wiring splice (See Figure F-1).

Splice wiring and remove tags.

Bolt section in place.

Reinstall flex connector to ice protection.

Install brush baffle assembly, support and remove sling.

Install opposite stabilizer and remove support.

C. Elevator Installation

Sling elevators into position.

Install nuts on eyebolts at hinge.

Install bolts attaching elevator to inboard hinge fitting.

Reconnect torque tube to tab drive.

Reconnect cranks to dampers and trim tab stop bolts.

Button up access doors.

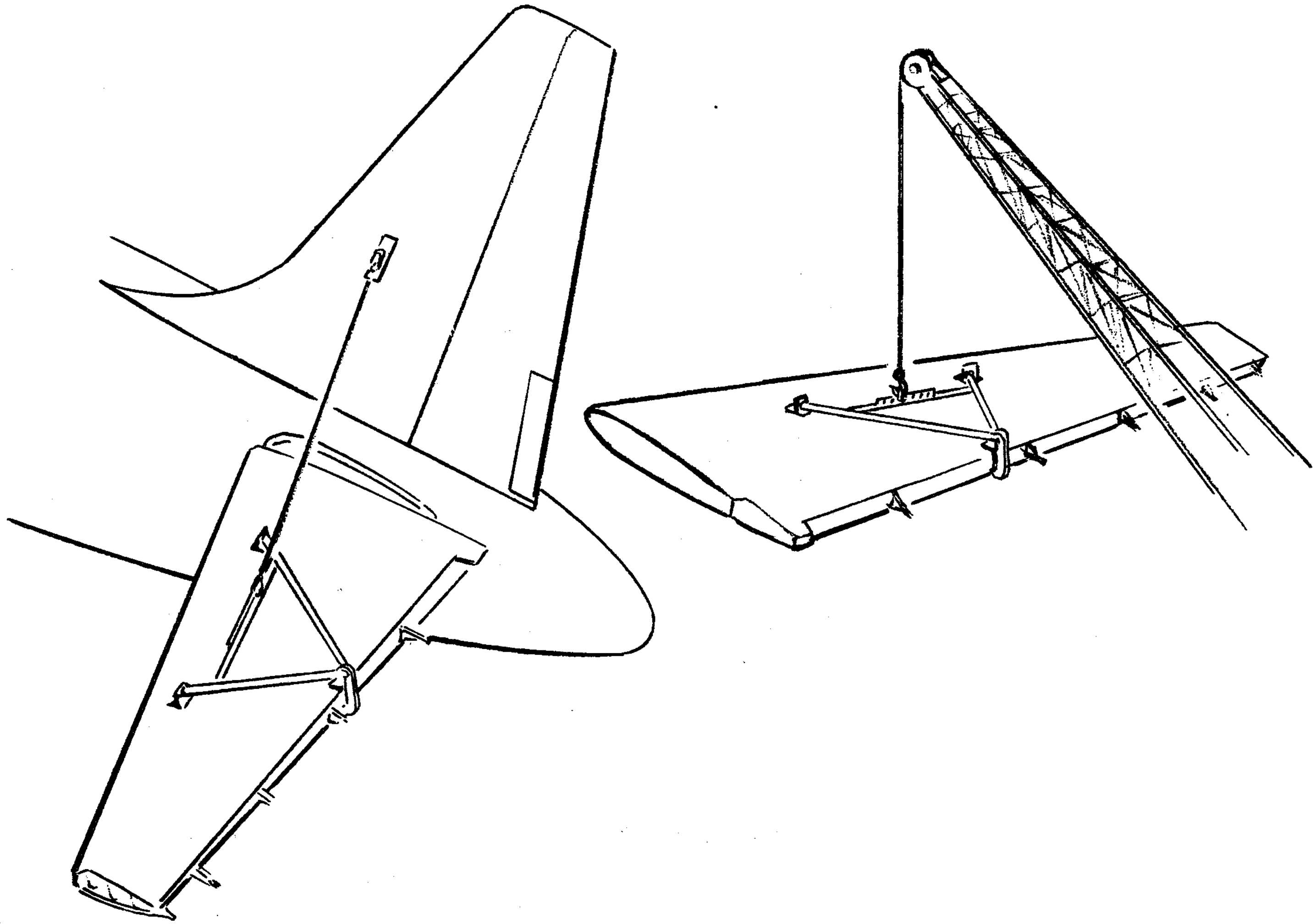


FIGURE F-1. DC-8 HORIZONTAL STABILIZER SLING SUPPORT

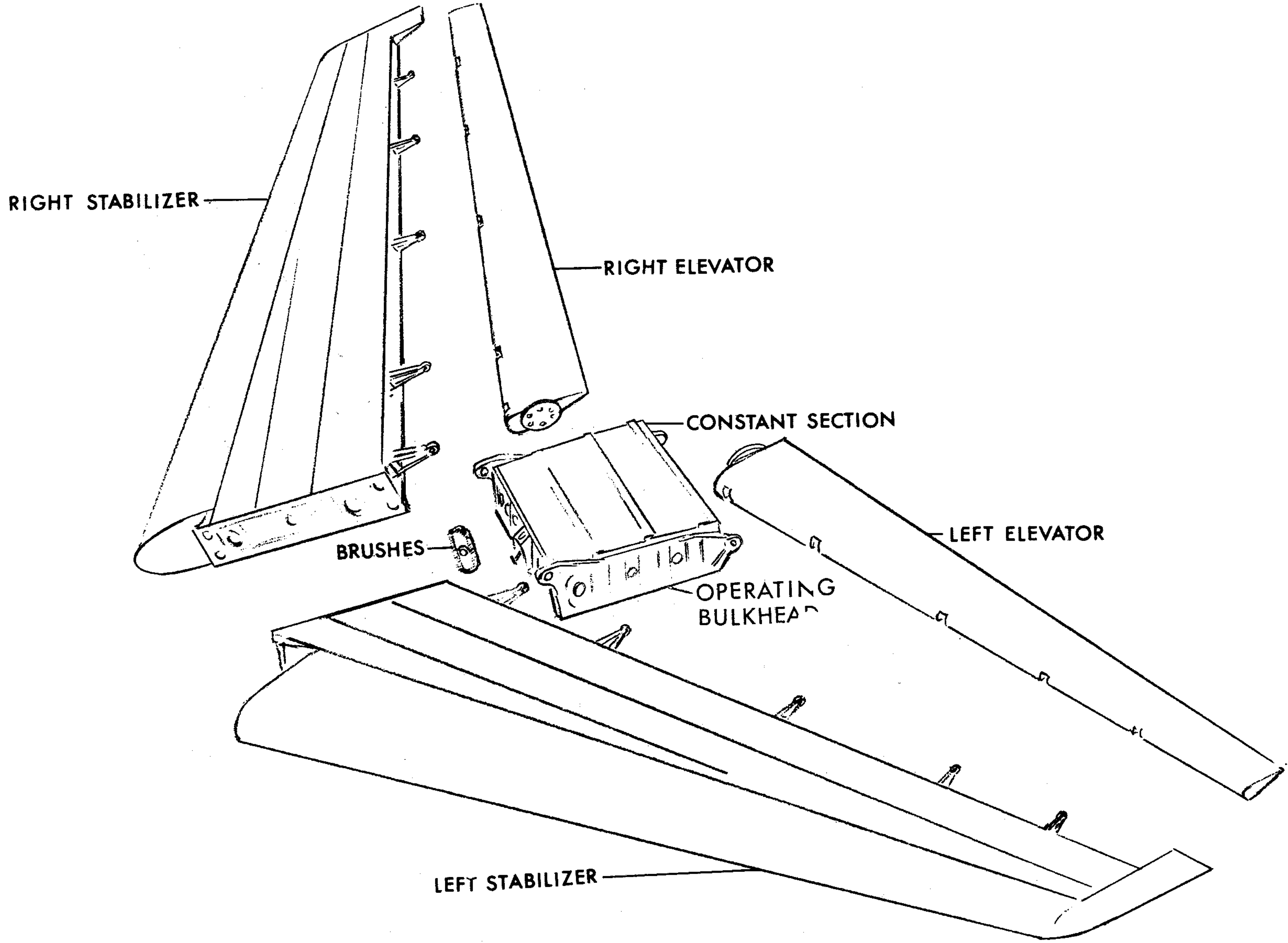


FIGURE F-2. DC-8 HORIZONTAL STABILIZER BREAKDOWN

NOTE: OPERATING BULKHEADS NOT SHOWN

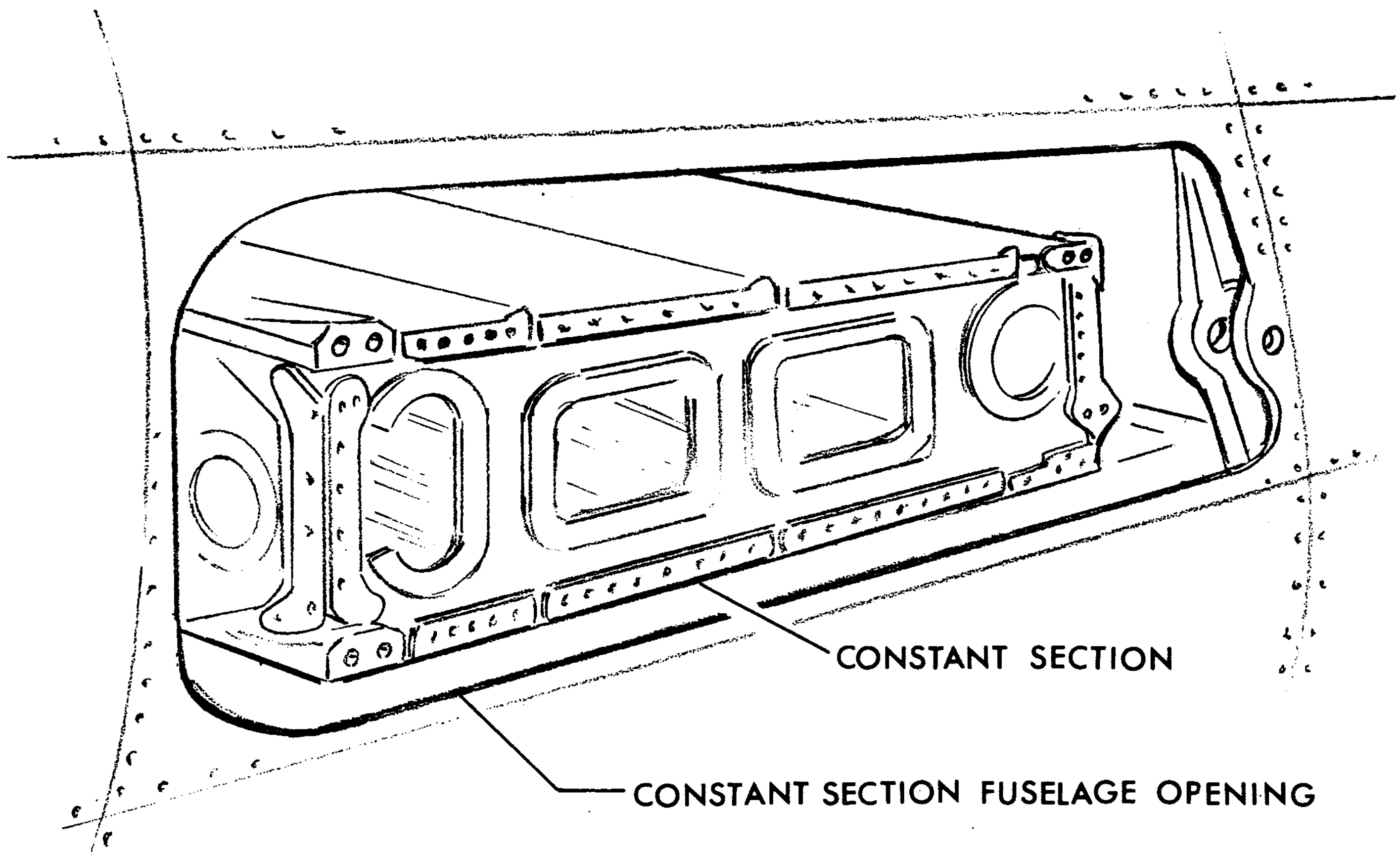


FIGURE F-3. DC-8 DISCONNECTED CONSTANT SECTION

APPENDIX G

COST-BENEFIT ANALYSES

I. BASIC APPROACH

Cost methodology studies were conducted along two separate but interrelated efforts. One effort was directed to the modification and subsequent utilization of an existing methodology for estimating the development and production costs of aluminum technology airframes (base case). This can provide manufacturing costs for the different categories of components, eg; fuselage, control surfaces, and floors which could then be used to determine the relative costs of utilizing composites in aircraft.

The second effort was directed toward a search for, and the preliminary development of a methodology which explicitly treats the cost uncertainties (design, materials, process and manufacturing) inherent in an emerging technology such as hybrid composite structures. Investigations made during this program indicated that additional effort would be required to adequately develop a methodology applicable to the design, analysis, test, and manufacture of hybrid composite structures.

These activities represent the preliminary steps toward the development of a general methodology for examining, at the advanced design level, the cost impact of composite applications ranging from reinforcement to substitution and original design.

II. PRODUCTION AND DEVELOPMENT COST ANALYSES

The production and development cost equations which are a part of a modified Rand Corporation methodology are identified[1]. Base case costs for initial and sustaining engineering, development support, flight test operations, initial and sustaining tooling, manufacturing labor, quality control, and manufacturing material are discussed.

[1]Levenson, G. S. and Barro, S. M., "Cost Estimating Relationships for Aircraft Airframes", Rand Corporation, RM-4845-PR (Abridged), May 1966.

A. Initial and Sustaining Engineering Costs

Total engineering costs were divided into initial and sustaining engineering. Initial engineering costs include the cost of engineering hours expended prior to the completion of the first airframe. Sustaining engineering costs account for the engineering hours which are a function of the quantity of aircraft produced. These are the recurring engineering costs over the production run.

The initial engineering hours, E_I, are estimated by the equation

$$E_I = .0633\ A^{.785} S^{1.428}$$

where:

S = Maximum design speed (kn)

A = AMPR Wt. (Roughly corresponds to DAC cost weight)

The sustaining engineering hours E_s, are estimated by the equation

$$E_s = E_I(N^{0.20} - 1)$$

where:

N = Cumulative quantity of airframes produced

B. Development Support Costs

Development support costs consist of the non-recurring manufacturing effort in support of engineering during the aircraft development phase. This includes the cost of manufacturing labor and material for mockups, test parts, and static test items, but excludes the flight test aircraft required for airframe design and development work.

Development support costs, D, are estimated as dollars per initial engineering hours as follows:

$$D = 17.00\ E_I$$

C. Flight Test Operations Costs

Flight test operations costs include all charges incurred by the contractor for flight tests, excepting the cost of the flight aircraft.

Flight test operations costs are estimated from the following equation.

$$F = .151\ A^{.93} S^{.98} n^{1.32}$$

where:

F = Flight test costs

n = Number of flight test aircraft

S = Maximum speed (kn)

A = AMPR Wt. (lbs)

D. Initial and Sustaining Tooling Costs

These cost equations cover the hours charged for tool design, tool planning, tool fabrication, production test equipment, checkout and maintenance of tooling, changes, and production planning.

The initial tooling hours, T_I, are estimated by the following equation:

$$T_I = .027\, A^{.99} S^{1.21} R^{0.4}$$

where:

R = Production rate in airframes per month

A = AMPR Wt. (lbs)

S = Maximum design speed (kn)

The sustaining tooling hours, T_s, are estimated by the following equation:

$$T_s = T_I\,(N^{0.14} - 1)$$

where:

N = Cumulative quantity of airframes produced

To convert tooling hour estimates into tooling cost, the hours are multiplied by a composite rate which includes direct labor, overhead and material costs.

E. Manufacturing Labor Costs

Manufacturing labor costs include those hours necessary to machine, process, fabricate, and assemble the major structure of the aircraft, and to install purchased parts, government furnished equipment (GFE) and offsite manufactured assemblies.

The manufacturing labor hours are estimated in the following steps:

- o The number of hours required to produce the 100th unit, $H_u(100)$ is

$$H_u(100) = .525A^{.85}S^{.54}$$

where:

A = AMPR Wt. (lbs)

S = Maximum design speed (kn)

- o Values of unit hours at other quantities, $H_u(N)$, are obtained from a learning curve equation with 75 percent learning assumed.*

$$H_u(N) = 6.76H_u(100)\ N^{-.415}$$

where:

N = Cumulative quantity of airframes produced

*The cost estimating relationships used in this study assume a 75 percent learning curve for manufacturing labor and an 89 percent learning curve for materials. These learning curves are generally represented by an exponential (log-linear) equation of the form $C = a\ X^{-b}$. The exponent -b, is 0.415 for a 75 percent learning curve and 0.168 for an 89 percent learning curve.

F. Quality Control Costs

Quality control costs consist of the hours expended in inspecting fabricated and purchased parts, sub-assemblies, and final installations and assemblies, against material and process standards, drawings, and specifications. These costs were estimated at 14 percent of total manufacturing labor costs.

G. Manufacturing Material Costs

Manufacturing material costs include the costs of raw materials and purchased parts.

Costs of avionics and engines are not included in material costs, nor are the non-recurring or development costs of purchased items which are covered in the development support costs.

The manufacturing material costs are computed using equations similar to those used for manufacturing labor. The material costs for the 100th airframe, $M_u(100)$, is obtained from:

$$M_u(100) = .306A^{.851}S^{.873}$$

where:

A = AMPR Wt. (lbs)

S = Maximum design speed (kn)

Unit material costs at other quantities, $M_u(N)$, were obtained from an 89 percent learning curve:

$$M_u(N) = 2.17M_u(100)N^{-0.168}$$

Unit material costs at any quantity may be converted to cumulative average cost by applying the ratio of cumulative average to unit cost from an 89 percent learning curve. For the 15th unit the ratio of cumulative average to unit cost is 0.738/0/634 = 1.16, so the cumulative average cost at unit 15 would be $1.16M_u(15)$.

The cumulative average cost times the number of airframes being considered equals the total material cost. The total material cost for n test airframes would be n times the cumulative average cost. The total material cost of N production airframes would be (N + n) times the cumulative average cost at the (N + n)th unit, minus the total material cost of the n test airframes.

III. COST ESTIMATING METHODS

Three cost estimating methods were considered, single-point, high and low costs, and probabilistic. It appears at this time that the single-point method would be the most feasible for the requirements of this program.

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