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A Study on the Utilization of Advanced Composites in Commercial Aircraft Wing Structure

Final Report

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Contract NAS1-15004
July 1978

NASA

National Aeronautics and Space Administration

Langley Research Center
Hampton, Virginia 23665

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A STUDY ON THE UTILIZATION OF ADVANCED COMPOSITES IN COMMERCIAL AIRCRAFT WING STRUCTURE

FINAL REPORT

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Aircraft Energy Efficiency Program

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by

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Long Beach, California

PREFACE

This final report was prepared by the Douglas Aircraft Company, McDonnell Douglas Corporation, under NASA Contract NAS1-15004, Study on Utilization of Advanced Composites in Commercial Aircraft Wing Structures. The study was conducted as part of the Composite Structures Element of the NASA Aircraft Energy Efficiency (ACEE) Program. The study program was monitored by Herman Bohon, ACEE Program Office, Langley Research Center. D. J. Watts was the Douglas Project Manager.

Principal Douglas contributors to this project were:

- C. Y. Kam – Composite Structures Technology
- F. M. Wright – Structural Design
- M. Platte – Cost Analysis
- J. Kung – Avionics
- A. Richter – Test
- R. Palmer – Materials and Producibility
- R. Hartunian – Manufacturing
- D. Retrum – Program Administration

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SUMMARY

A study was conducted to define the technology and data needed to support the introduction of advanced composite materials in the wing structure of future production aircraft. In the course of the study, discussions were held with key personnel from airlines, the Federal Aviation Administration (FAA), and Douglas Aircraft Company management. Their participation ensured that the study findings are representative for a broad segment of the commercial transport aircraft community.

The study accomplished the following:

- Definition of acceptance factors
- Identification of technology issues
- Evaluation of six candidate wing structures
- Evaluation of five program options
- Definition of a composite wing technology development plan
- Identification of full-scale tests
- Estimation of program costs for the total development plan
- Forecast of future utilization of composites in commercial transport aircraft
- Identification of critical technologies for timely program planning.

A comprehensive list of acceptance factors was formulated for the manufacturer, airlines, and FAA. Concurrence with the factors listed has been received from cognizant personnel from each of the three sectors.

A set of 24 issues was derived from the acceptance factors to form the basis for a technology assessment. Each issue was examined to determine which technological or economic problems must be resolved by a composite wing technology program. Recognition was given to probable contributions to the technology by other composite programs in Government and industry so that they need not be repeated in a composite wing technology program.

Eight of the issues were classified as key issues:

- Durability
- Damage tolerance
- Crashworthiness
- Repair of major damage
- Lightning protection
- Molding methods
- Nondestructive inspection methods
- Large-scale tools.

These key issues are addressed in the development plan. Other issues will be addressed in the process of conducting a composite wing technology program, as defined herein.

Six candidate wing structures were evaluated for the baseline wing component. The DC-9-32 wing was selected on the basis of size, availability for commercial transport, availability of design data, and the presence of design features that cover a realistic and comprehensive range of composite wing technology.

Five program options were formulated. Based on the technology assessment, it was determined that a common thread existed for all options:

- Design synthesis
- Development tests
- Manufacturing technology
- Operational technology
- Detail design.

The program options vary only in the size and quantity of full-scale hardware produced, in the amount of verification testing conducted, and in the scope of flight development and flight evaluation. Details of the program option which was selected for the composite wing technology program are defined in the development plan.

A conceptual composite wing box was designed which accounted for interface with adjoining structure and aircraft subsystems. A 28-percent weight saving was realized for this design compared to the existing metal wing design.

A development plan has been defined for the DC-9-32 composite wing box. Development activities are divided into six phases:

Phase I	Preliminary Design
Phase II	Detail Design
Phase III	Manufacturing
Phase IV	Full-Scale Tests
Phase V	Flight Development
Phase VI	Flight Evaluation.

Full-scale semispan composite wing box hardware will be fabricated rather than full-span hardware. This approach will eliminate the need for opposite-hand tools and reduce the quantity of hardware produced, which will lower costs.

The following full-scale tests are specified:

- Static ultimate
- Durability and damage tolerance
- Crashworthiness
- Repair of major damage
- Vibration.

The production facilities and equipment forecast for composite wing structures was made with the awareness that primary wing structure would be preceded by secondary and medium primary structure utilization throughout the airframe. A total floor space buildup to 55,742 square meters (600,000 square feet) dedicated to composite structures would be required to produce a production airplane with composite primary wing structure.

Total program costs for a composite wing development program are estimated at $74.9 million (ROM) based on 1978 dollars. Of this total, 32 percent is allocated to the Phase I preliminary design and 45 percent to the Phase III manufacturing (includes tooling). The remaining 23 percent is approximately evenly divided among the other four phases.

A road map is presented for utilization of composite structures on future Douglas production commercial transport aircraft. This road map reveals Company plans for a logical progression to a composite wing box on a short-haul transport planned for first production delivery in 1990.

The study concludes that it is highly improbable that a production commitment will be made until a comprehensive composite wing development program has produced data and technology sufficient to resolve the economic, programmatic, and technological risks identified by this study.

If the study objective of a composite wing box on a 1985-1990 production aircraft is to be realized, activity must be started in 1979 on the following key issues for which data are needed at the start of the preliminary design (Phase I) or which must be started early due to the time required to produce data and develop technology:

- Repair of major damage
- Impact damage (included in durability issue)
- Damage tolerance design studies and tests
- Innovative molding methods
- Tooling methods for large composite structures
- Lightning protection.

Activity on the remainder of the durability issue and the other two key issues of crashworthiness and nondestructive inspection methods can be started later in Phase I since basic data for these technologies are available to support early preliminary design tasks.

SECTION 1
INTRODUCTION

The overall wing study objectives are to study and plan the effort required by manufacturers of commercial transport aircraft to accomplish the transition from current conventional materials and practices to extensive use of advanced composites in wings of aircraft that will enter service in the 1985-1990 time period.

Specific wing study objectives are to define the technology and data needed to support an aircraft manufacturer's commitment to utilize composite primary wing structure in future production aircraft and to develop plans for a composite wing technology program which will provide the needed technology and data.

Figure 1-1 presents a task flow diagram to achieve study objectives.

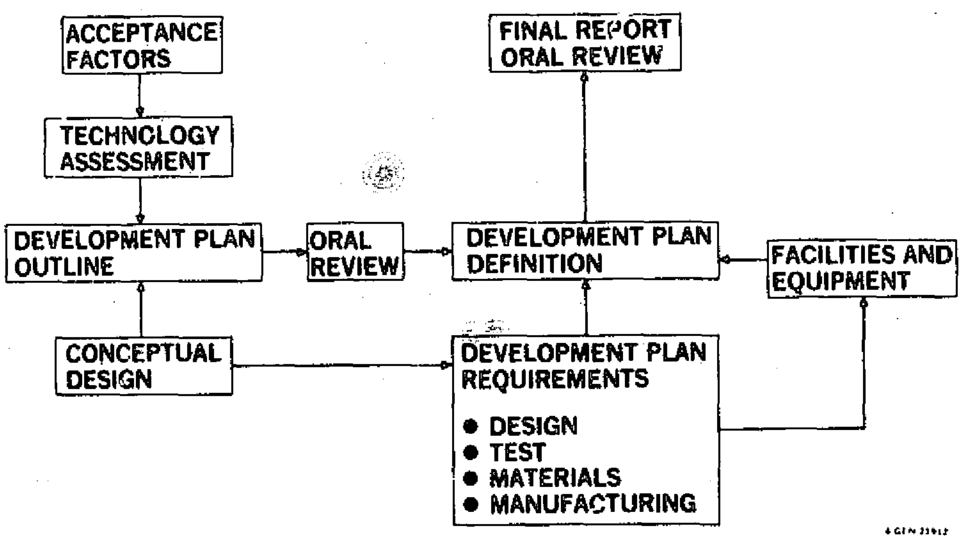

FIGURE 1-1. COMPOSITE WING STUDY FLOW DIAGRAM

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SECTION 2
ACCEPTANCE FACTORS

A manufacturer's decision to utilize composite wing box structure in commercial transport aircraft will be strongly influenced by the attitude of the airline operators and the FAA. Each considers many of the same factors, most notably structural integrity. Factors related to cost are of primary concern to the manufacturers and the airlines.

The acceptance factors listed in Table 2-1 form the basis for the technology assessment to identify those issues which must be resolved to gain airline acceptance, approval for airworthiness, and a manufacturer's commitment to production of composite wing box structure.

AIRLINE ACCEPTANCE FACTORS

During the past 2 years, the airlines' attitude toward advanced composite structure has been changing from skepticism to a positive approach of wanting to learn more about the new materials and to prepare for their eventual introduction as production structural materials. To quote one airline Engineering Vice President, "It seems inevitable that we are going to have to take advantage of these new materials to reduce fuel consumption. If this is the case, then we must start the gradual introduction now on secondary structure so that we can be prepared for more extensive utilization in the future." Toward this objective, a number of airlines are currently flying advanced composite structure in the following components:

- DC-10 rudder
- DC-10 vertical stabilizer trailing edge panel
- DC-10 pylon fairing
- B737 spoiler
- B707 flap vane
- L-1011 aileron fairings
- DC-9 nacelle cowl doors.

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TABLE 2-1
ACCEPTANCE SUMMARY

	MANUFACTURER	AIRLINES	FAA
STRUCTURAL INTEGRITY FACTORS	X	X	X
1. MATERIAL AND FABRICATION	X	X	X
2. STATIC STRENGTH	X	X	X
3. FATIGUE/DAMAGE TOLERANCE	X	X	X
4. CRASHWORTHINESS	X	X	X
5. FLAMMABILITY	X	X	X
6. LIGHTNING PROTECTION	X	X	X
7. PROTECTION OF STRUCTURE	X	X	X
8. QUALITY CONTROL	X	X	X
9. REPAIR	X	X	X
10. FABRICATION METHODS	X	X	X
OPERATIONAL FACTORS			
11. RELIABILITY		X	
12. MAINTAINABILITY		X	
13. INSPECTABILITY		X	
14. REPAIRABILITY		X	
ECONOMIC FACTORS			
15. ACQUISITION COSTS		X	
16. LIFE-CYCLE COSTS		X	
17. WARRANTIES		X	
18. FACILITIES	X	X	
19. EQUIPMENT	X	X	
20. PRODUCTION COSTS	X	X	
PROGRAMMATIC RISK FACTORS			
21. DESIGN DATA	X		
22. PRODUCIBILITY DATA	X		
23. SCHEDULE DATA	X	X	
24. COST DATA	X	X	
25. STAFF EXPERIENCE	X	X	
26. AIRLINE ACCEPTANCE	X		
27. FAA ACCEPTANCE	X	X	

Douglas has contacted a large number of airline officials to determine what they feel is necessary before they could accept an advanced composite primary wing box structure. The contacts have been made by several Company departments.

The airlines' acceptance of composite structure appears to depend on assurance of structural integrity and cost. The general consensus is that if the manufacturer and the FAA are satisfied with the level of structural integrity of composite structure, the material can be proven acceptable to the airlines.

On a cost basis, the airlines are concerned with acquisition costs, maintenance costs, inspection costs, special equipment and facilities, out-of-service time, and replacement costs. An excellent in-depth discussion of this subject is provided in Reference 1.

Cost-oriented airline acceptance factors have been identified under the following general headings:

- Reliability — Unscheduled time out of service is an extremely high cost factor. Data must be provided to the airlines to assure dispatch reliability equivalent to that of conventional structural materials.

- Maintainability — Maintenance and inspection costs fall in this category. Airlines will expect to see evidence that composite wing box structures can be maintained as readily as conventional aluminum structure. Inspection is a major concern. The airlines need to know what equipment they must acquire for inspections, and must train personnel to conduct the inspections. The manufacturer must supply them with inspection methods and FAA-approved intervals.

- Durability — Durability in a service environment must be proven. Chaste laboratory tests must be supplemented with environemtal exposure tests (heat, cold, ice, slush, skydrol, fuel, etc.) to provide credible evidence of durability.

- Repairability — The airlines will not accept structure unless workable repair schemes have been demonstrated. Repair of major damage of composite structure is the foremost concern. Facilities and equipment must be available at a major repair depot, and cost-effective repairs must be accomplished in the same time span as for aluminum wing structure repairs. The airlines currently consider the repairs of major damage as a major risk item, and evidence must be presented that major damage to composite wing box structure can be repaired without incurring time-out-of-service costs greater than the equivalent costs for metal structure.

- Warranty — The airlines expect the manufacturers to provide the same level of warranty as for aluminum structure. During recent years, warranty coverage for commercial transport aircraft has escalated for conventional structure due to the improvement in structural designs on successive new models. The airlines feel that this upward trend should not be interrupted by the introduction of composite structures.

- Tangible Benefits — The airlines must be presented with evidence that they will benefit financially through the utilization of composite materials. There is a risk factor associated with new designs and the composite wing box will not be wanted unless a payoff is apparent. This should be expressed in terms of reduced fuel costs, higher payload capability, reduced maintenance costs, and lower replacement costs.

- Acquisition — The original equipment costs must be reasonable to allow the aircraft to be competitively priced.

FAA ACCEPTANCE FACTORS

FAA acceptance factors for advanced composite materials have been well defined. Guidelines have been drafted and FAA Advisory Circular AC20-107, entitled "Certification Guidelines for Civil Composite Aircraft Structures," has been published (Reference 2). These guidelines are considered acceptable to the FAA for showing compliance with certification requirements of civil composite structure. It is expected that the guidelines will be modified periodically to reflect advances in technology.

Table 2-2 lists the general topics for which guideline material is provided and includes an index to applicable FAR requirements.

TABLE 2-2
CERTIFICATION GUIDELINES FOR CIVIL COMPOSITE WING AIRCRAFT STRUCTURES

• MATERIAL ALLOWABLES	– FAR 25.603, 25.613 AND 25.615
• PROOF OF STRUCTURE – STATIC	– FAR 25.305 AND 25.307(a)
• PROOF OF STRUCTURE – FATIGUE/DAMAGE TOLERANCE	– FAR 25.571 (PROPOSED NEW AND APPENDIX)
• CRASHWORTHINESS	– FAR 25.561, 25.721, 25.801(b) (e), AND 25.963(d)
• FLAMMABILITY	– FAR 25.863(b) (5), 25.867, 25.1191 AND 25.1193
• LIGHTNING PROTECTION	– FAR 25.581
• PROTECTION OF STRUCTURE	– FAR 25.609
• QUALITY CONTROL	– FAR 21.143
• REPAIR	– FAR 121.367(a) AND FAR 43.13(a)
• FABRICATION METHODS	– FAR 25.603 AND 25.605

MANUFACTURER ACCEPTANCE FACTORS

The decision to produce composite wing box structure will be made at the highest management level. Both technical and economic factors will be assessed in the evaluation process.

The first factor to be addressed deals with the motive for the utilization of composite wing box structures. Manufacturing cost data and the increase in aircraft performance due to weight savings will be assessed to determine if the benefits outweigh the risk of a new venture.

The second factor to be considered is the proven structural integrity of composite wing box structure. Management must be presented with evidence that the strength, reliability, durability, damage tolerance, etc. of composite structure have been demonstrated to be sufficient to satisfy concerns of the manufacturer and airline with function and safety.

Additional evidence with respect to maintenance, inspection, and repairability is required to assure that the structural integrity of the composite wing box can be maintained throughout the life of the aircraft. The evidence should also demonstrate that compliance with FAA regulations can be accomplished without undue delays in meeting the schedule or unanticipated expenses.

Once the benefit motive and technical feasibility have been established, the manufacturer must have the capability to design and produce the composite wing box structure. Management must be confident that a low-weight design can be created, tools built, certifiable components manufactured, and the airplane certified in accordance with delivery schedules and within the predicted costs to ensure that the motives for utilization of composite structure have not been compromised.

The capability required covers a broad spectrum in the field of composite structures:

- An engineering data base for composite structures must be available to support early design tasks and to minimize the development costs.

- Manufacturing technology must be developed to provide low-risk, cost-effective manufacturing methods. The manufacturing methods to be used must be established during the preliminary design; a deficiency in manufacturing technology will affect cost, schedules, and structural integrity.

- Facilities and equipment must exist or must be provided as required to meet delivery schedules. The facilities and equipment utilized for a production composite wing box structure influences the selection of manufacturing method, which in turn has a significant impact on manufacturing cost.

- A staff of design engineers, materials and process engineers, manufacturing engineers, quality assurance personnel, and production workers must be available with the necessary experience in composite structures to serve as the cadre for training and supervising the expanded staff required for production of composite wing box structure.

SECTION 3
TECHNOLOGY ASSESSMENT

The acceptance factors have been translated into a set of issues which need to be assessed as a prelude to defining the contents of a composite wing technology program. The issues can be categorized into four basic groups, as shown in Figure 3-1.

In making the assessment, each issue has been examined to determine what additional technology and data are needed to promote acceptance of composite primary wing structure. It is assumed that all technology and data required to design, manufacture, and certify the earlier NASA ACEE secondary and medium primary structures will be available. For example, the secondary and medium primary structures utilize more thin-gauge panels, and the need for postbuckling strength allowables is greater than for strain-critical wing cover panels. Therefore, although a knowledge of postbuckling strength is desirable for minimum-weight wing structure, it is assumed the technology will be available and is not addressed in the wing technology assessment. Contributions from other Government, industry, and in-house projects have also been anticipated to minimize the composite wing technology program costs. Information on some aircraft composite structure programs which was used as reference material for the state-of-the-art assessment is presented in Table 3-1.

Twenty-four issues have been selected for the technology assessment, as shown in Figure 3-2. Of these, eight have been classified as key issues since favorable resolution is essential to the timely production of composite wing structure and specific technology development plans for their resolution must be included in the overall development program.

STRUCTURAL TECHNOLOGY ISSUES

Five of the seven structural technology issues shown in Figure 3-2 are classified as key issues. Composite wing structure must be produced with the same level of structural integrity as for conventional aircraft wing structure and evidence of this must be provided as a condition of acceptance. Technical data must be generated during the design synthesis of a prototype

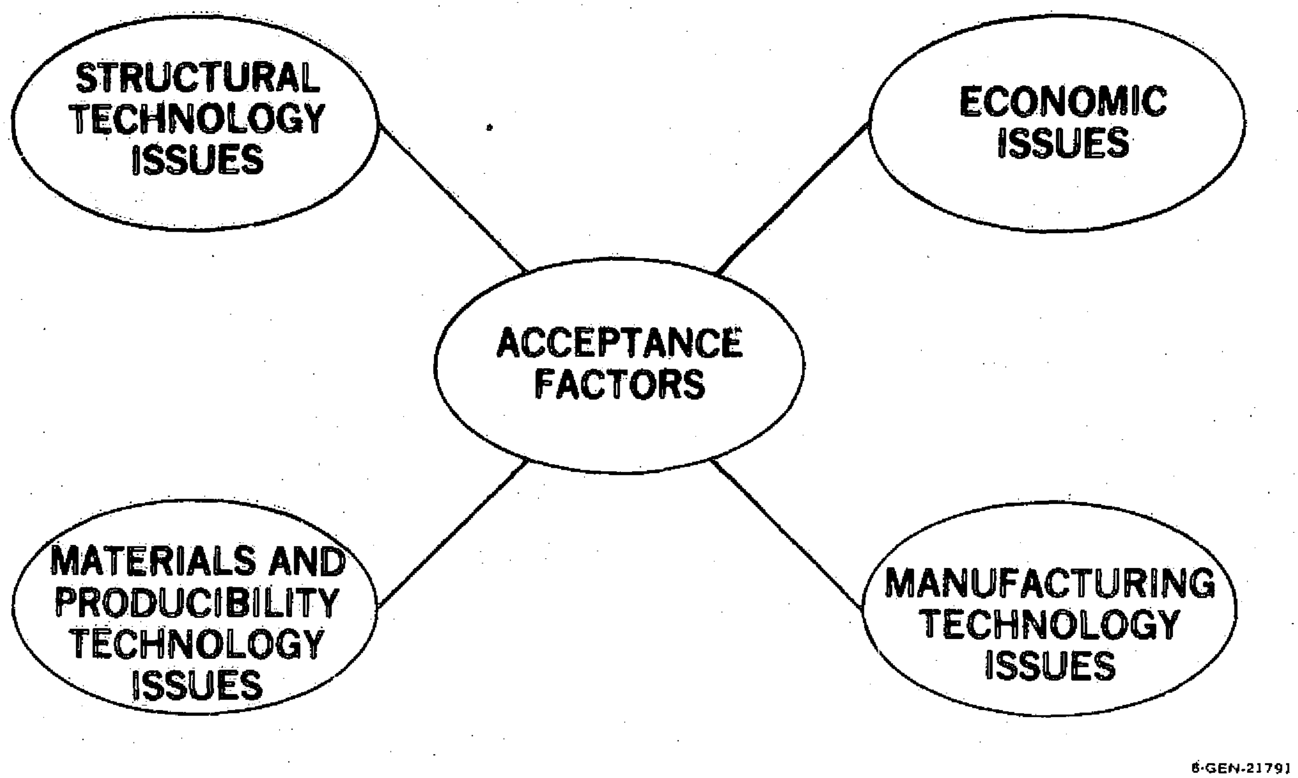

FIGURE 3-1. KEY ISSUE GROUPS

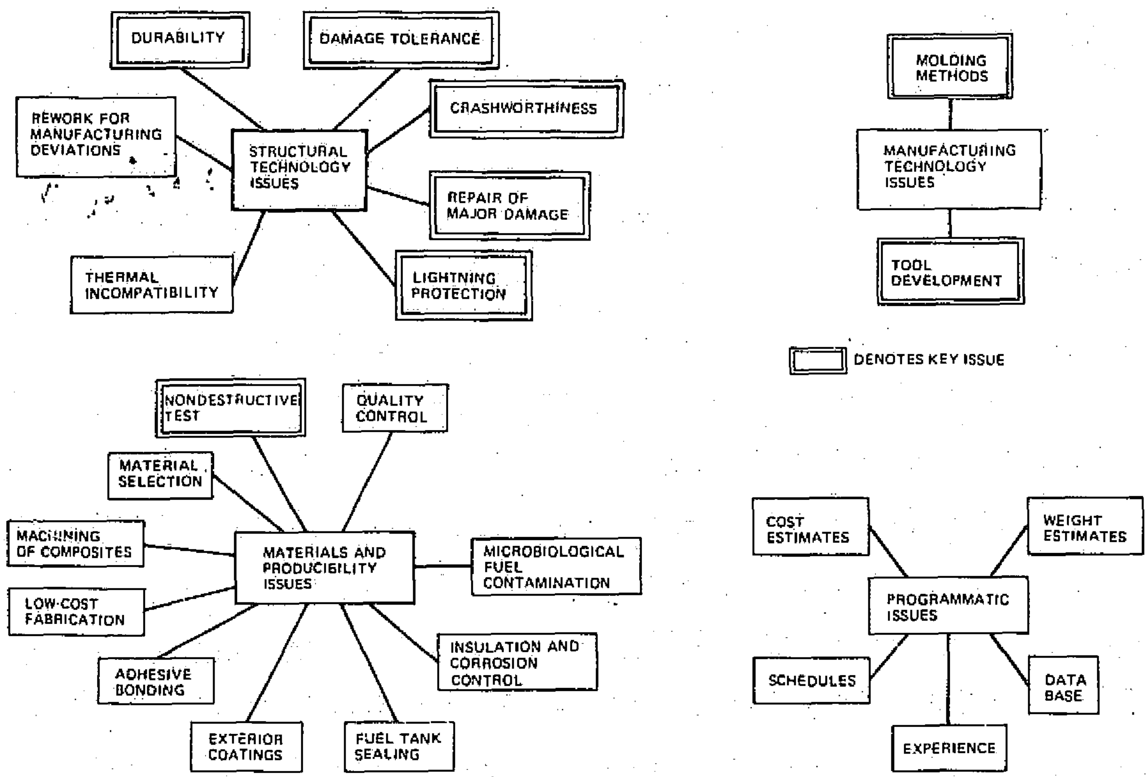

FIGURE 3-2. BASIS FOR TECHNOLOGY ASSESSMENT

TABLE 3-1

SOME ADVANCED COMPOSITES APPLICATIONS IN AIRCRAFT STRUCTURES

PROGRAM	STRUCTURAL CONCEPT	MANUFACTURING CONCEPT	COMMENTS
DC-10 AFT RUDDER DOUGLAS AIRCRAFT COMPANY – MDC ACEE PROGRAM	• BOX BEAM • MULTIRIB • THIN SKIN	• INTEGRAL ONE PIECE • USING TRAPPED RUBBER PRESSURING SYSTEM	• REQUIRES ONE EXPENSIVE FEMALE MOLDING TOOL – ECONOMIC FOR MASS PRODUCTION • PREFORMING OF PARTS • ASSEMBLY OF PARTS AND SILICONE RUBBER IN TOOL • ELIMINATES MECHANICAL FASTENERS AND COSTLY ASSEMBLY • MAXIMIZES WEIGHT SAVING • ACCESS HOLES ALLOW INSPECTION AND REPAIR • RUDDERS PRESENTLY FLYING ON DC-10 AIRCRAFT ATP JAN 1974 – PRODUCTION CONTINUING "ADVANCED COMPOSITE RUDDERS FOR DC-10 AIRCRAFT – DESIGN MANUFACTURING AND GROUND TESTS" REPORT NO. NASA CR-145068 NASA LANGLEY RESEARCH CENTER, HAMPTON, VIRGINIA, CONTRACT NO. NAS1-14724
DC-10 VERTICAL STABILIZER DOUGLAS AIRCRAFT COMPANY – MDC ACEE PROGRAM	• MULTISPAR • SINEWAVE SHEAR-WEBS • SINEWAVE RIBS • SANDWICH SKINS	• SPARS, RIBS, AND SKINS SEPARATELY FABRICATED • SKINS ASSEMBLED TO SUB-STRUCTURE WITH MECHANICAL FASTENERS	• LOW MANUFACTURING RISK • HIGH TOOLING COST • ASSEMBLY LESS COSTLY THAN METAL VERSION BECAUSE OF FEWER PARTS • HEAVIER THAN A COCURED INTEGRAL DESIGN • EASIER TO INSPECT AND REPAIR ATP APRIL 1977 – DEVELOPMENT IN PROGRESS, "ADVANCED COMPOSITE VERTICAL STABILIZER FOR DC-10 TRANSPORT AIRCRAFT" REPORT NO. ACEE-03-PR-7177, 7240, AND 8232, 20 JAN 1978 NASA, LANGLEY RESEARCH CENTER, HAMPTON, VIRGINIA, CONTRACT NO. NAS1-14869
BOEING 737 HORIZONTAL STABILIZER ACEE PROGRAM	• BOX BEAM • SKIN WITH I-SECTION STRINGERS	• FABRICATION OF SEPARATE SPARS AND RIBS • SKIN-STRINGERS COCURED • MECHANICAL FASTENERS USED FOR ASSEMBLY	• LOW MANUFACTURING RISK • HIGH TOOLING COST • ASSEMBLY LESS COSTLY THAN METAL VERSION BECAUSE OF FEWER PARTS • EASIER TO INSPECT ANDREPAIR ATP JULY 1977 – DEVELOPMENT IN PROGRESS, "ADVANCED COMPOSITE STABILIZER FOR BOEING 737 AIRCRAFT" FIRST QUARTERLY PROGRESS REPORT, 18 JULY 77 – 18 OCTOBER 77 NASA, LANGLEY RESEARCH CENTER, HAMPTON, VIRGINIA, CONTRACT NO. NAS1-15025

TABLE 3-1

SOME ADVANCED COMPOSITES APPLICATIONS IN AIRCRAFT STRUCTURES (CONT)

PROGRAM	STRUCTURAL CONCEPT	MANUFACTURING CONCEPT	COMMENTS
BOEING 727 ELEVATOR ACEE PROGRAM	• BOX BEAM • SANDWICH RIBS • SANDWICH SKINS • MECHANICAL FASTENERS USED FOR ASSEMBLY	• FABRICATION OF SEPARATE SPARS, RIBS, AND SKINS	• COMMENTS MADE FOR BOEING 737 HORIZONTAL STABILIZER APPLY HERE ATP MAY 1977 – DEVELOPMENT IN PROGRESS "ADVANCED COMPOSITE ELEVATOR FOR BOEING 727 AIRCRAFT" FIRST QUARTERLY PROGRESS REPORT, 24 MAY 77 – 22 AUGUST 77 NASA, LANGLEY RESEARCH CENTER, HAMPTON, VIRGINIA, CONTRACT NO. NAS1-14952
L-1011 VERTICAL STABILIZER LOCKHEED AIRCRAFT COMPANY BURBANK, CALIFORNIA ACEE PROGRAM	• BOX BEAM WITH RIBS • RIBS – TRUSS-TYPE AND PANEL STIFFENED-TYPE • SKINS STIFFENED WITH HOT STRINGERS	• SEPARATE FABRICATION OF SPARS, RIBS, SKIN-STIFFENERS • HAT STIFFENERS COCURED WITH SKIN • SPARS ARE INTEGRALLY STIFFENED • ASSEMBLED WITH MECHANICAL FASTENERS	• LOW MANUFACTURING RISK • HIGH TOOLING COST, ESPECIALLY WITH MULTIPLE-RIB DESIGNS • ASSEMBLY LESS COSTLY THAN METAL VERSION BECAUSE OF FEWER PARTS • LIGHTER THAN METAL VERSION • EASE OF INSPECTION AND REPAIR ATP AUGUST 1975 – DEVELOPMENT IN PROGRESS. "ADVANCED MANUFACTURING DEVELOPMENT OF A COMPOSITE EMPENNAGE COMPONENT FOR L-1011 AIRCRAFT," LR 28326 – 14 OCTOBER 77, QUARTERLY TECHNICAL REPORT NO. 7 NASA, LANGLEY RESEARCH CENTER, HAMPTON, VIRGINIA, CONTRACT NO. NAS1-14000
L-1011 INBOARD AILERON LOCKHEED ACEE PROGRAM	• GRAPHITE/EPOXY FRONT SPAR • GRAPHITE/EPOXY RIBS • GRAPHITE/EPOXY FACE SHEETS WITH SYNTACTIC EPOXY CORE FOR COVER SKIN • ALUMINUM REAR SPAR • FIBERGLASS END FAIRINGS • KEVLAR TRAILING EDGE	• SEPARATE FABRICATION OF RIBS, SPARS, COVER SKIN • ASSEMBLED WITH MECHANICAL FASTENERS	• LOW MANUFACTURING RISK • COST EFFECTIVE DUE TO REDUCED PART COUNT • ESTIMATED 30% WEIGHT SAVINGS • LOWER COVER REMOVABLE FOR INSPECTION AND REPAIR ATP SEPTEMBER 1977 "ADVANCED COMPOSITE AILERON FOR L-1011 TRANSPORT AIRCRAFT" - QUARTERLY REPORT NO. 3, LR 28663, NASA, LANGLEY RESEARCH CENTER, HAMPTON, VA., CONTRACT NO. NAS1-15069, JULY 1978.
BOEING 737 TRANSPORT AIRCRAFT FLIGHT SPOILERS (REPLACING ALUMINUM SANDWICH SKIN WITH G/E SKINS)	• FULL-DEPTH SANDWICH G/E SKINS/ALUMINUM HONEYCOMB • SUBSTRUCTURE, SPARS ALUMINUM, FIBERGLASS END RIBS	• SKINS LAMINATED AND CURED IN AUTOCLAVE • SKIN SECONDARY BONDED TO CORE-SPAR-END RIBS, AS WELL AS BOLTED TO THE SPAR AND END RIBS • THE ALUMINUM SUB-STRUCTURE ASSEMBLED WITH MECHANICAL FASTENERS	• LOW MANUFACTURING RISK • TOOLING COST COMPARABLE TO METAL VERSION • ASSEMBLY COST ABOUT SAME AS METAL VERSION • C-SCAN INSPECTION FOR BOND AND DELAMINATION POSSIBLE • REPAIR NOT DIFFICULT ATP JUNE 1972 CONTRACT EXTENDED TO 1980 – IN PRODUCTION AND PRESENTLY FLYING "DEVELOPMENT MANUFACTURING AND TEST OF G/E COMPOSITE SPOILERS FOR FLIGHT SERVICE ON B737 TRANSPORT AIRCRAFT" – OCTOBER 1976 NASA, LANGLEY RESEARCH CENTER, HAMPTON, VIRGINIA, CONTRACT NO. NAS1-11668

TABLE 3-1

SOME ADVANCED COMPOSITES APPLICATIONS IN AIRCRAFT STRUCTURES (CONT)

PROGRAM	STRUCTURAL CONCEPT	MANUFACTURING CONCEPT	COMMENTS
AV-8B WING HARRIER AIRCRAFT MCDONNELL AIRCRAFT COMPANY – MDC	• MULTISPAR • RIB BULKHEADS • SEPARATE TOP AND BOTTOM SKINS	• FABRICATION OF SEPARATE SPAR AND RIBS • UPPER AND LOWER SKINS ARE EACH MONOLITHIC • ASSEMBLED THROUGH MECHANICAL FASTENERS	• LOW MANUFACTURING RISK • HIGH TOOLING COST • ASSEMBLY COST LESS THAN FOR METAL VERSION BECAUSE OF FEWER PARTS • EASIER TO INSPECT AND REPAIR ATP OCTOBER 1975 – DEVELOPMENT IN PROGRESS. "COMPOSITE WING DESIGN FOR ADVANCED DESIGN" REPORT NO. NADC-76249-30 NAVAL AIR DEVELOPMENT CENTER, WARMINSTER, PENN., CONTRACT NO. N62269-C-0424
F-15 WING EAGLE MCDONNELL AIRCRAFT COMPANY – MDC	• MULTISPAR • RIBS • SKINS BORON/EPOXY • SANDWICH WITH ALUMINUM CORE • STRINGERS HYBRID (BORON/GRAPHITE/EPOXY) • SPARS AND RIBS GRAPHITE/EPOXY SANDWICH SHEAR WEBS	• FABRICATION OF SEPARATE SPAR RIBS, SKINS, STRINGERS • STRINGERS BONDED TO THE SKINS • SKINS ATTACHED WITH MECHANICAL FASTENERS	• COMMENTS FOR HARRIER AIRCRAFT APPLY HERE, TOO. ATP MAY 1971 – DECEMBER 1974 R&D PROGRAM "F-15 COMPOSITE WING," VOLUMES I AND II, MAY 1975, AFML-TR-75-78 AFML, WRIGHT-PATTERSON AFB, OHIO, CONTRACT NO. F33615-71-C-11536
A-4 SKYHAWK HORIZONTAL STABILIZER DOUGLAS AIRCRAFT COMPANY – MDC	• MULTISPAR • RIB BULKHEADS • SPAR AND RIBS USE SANDWICH SHEAR WEBS	• SPAR, RIBS, AND SKIN SEPARATELY FABRICATED • SPAR AND RIBS BONDED TOGETHER AT CORNERS WITH G/E ANGLES • THIN SKIN MECHANICALLY FASTENED	• LOW MANUFACTURING RISK • HIGH TOOLING COST • ASSEMBLY LESS COSTLY THAN METAL VERSION BECAUSE OF FEWER PARTS • HEAVIER THAN A COCURED INTEGRAL DESIGN • EASIER TO INSPECT AND REPAIR ATP NOVEMBER 1969-1974 "DEVELOPMENT OF A GRAPHITE HORIZONTAL STABILIZER" FINAL REPORT – MDC J-6902 OR, NADC-76078-30 OR, ADA-023-767, MARCH 1976 NAVAL AIR DEVELOPMENT CENTER, WARMINSTER, PENN., CONTRACT NO. N00156-70-C-1321

TABLE 3-1

SOME ADVANCED COMPOSITES APPLICATIONS IN AIRCRAFT STRUCTURES (CONT)

PROGRAM	STRUCTURAL CONCEPT	MANUFACTURING CONCEPT	COMMENTS
ROCKWELL B-1 BOMBER HORIZONTAL STABILIZER STABILIZER MANUFACTURER GRUMMAN AEROSPACE CORP.	• MULTISPAR WITH FEW RIBS • SKIN WITH INTEGRAL STIFFENERS AND ALSO WITH BORON FILAMENTS	• FABRICATION OF SEPARATE SPARS AND RIBS • SKIN-STRINGERS INTEGRAL • MECHANCAL FASTENERS USED FOR ASSEMBLY	• COMMENTS MADE FOR BOEING 737 HORIZONTAL STABILIZER APPLY HERE "ADVANCED DEVELOPMENT OF CONCEPTUAL HARDWARE HORIZONTAL STABILIZER," ATP JULY 1973 – DECEMBER 1977 AIR FORCE FLIGHT DYNAMICS LABORATORY, DAYTON, OHIO, CONTRACT NO. F33615-73-C-5173
S-3A SPOILER LOCKHEED	• SANDWICH CONSTRUCTION WITH HRP HONEYCOMB CORE WITH G/E SKINS • HINGE FITTINGS BOLTED TO THREADED CORE INSERTS BOLTED INTO THE GLASS REINFORCED CORE	• SANDWICH COCURED • FITTINGS ATTACHED LATER	• LOW MANUFACTURING RISK • LOW TOOLING COST • LOW ASSEMBLY COST • HIGH STIFFNESS/WEIGHT RATIO • ACCESSIBLE TO INSPECTION AND REPAIR ATP 1974 – FLYING ON S-3A AIRCRAFT "S-3A GRAPHITE/EPOXY SPOILER FABRICATION OF TEN SHIPSETS AND DAMAGE REPAIR STUDY" NADS-76234-30, FINAL TECHNICAL REPORT, MAY 1976, NAVAL AIR DEVELOPMENT CENTER, WARMINSTER, PENN. CONTRACT NO. N62269-75-C-0428
B-1 FLAP FABRICATED BY NORTHROP CORPORATION	• FULL-DEPTH ALUMINUM HONEYCOMB WITH G/E SKINS • ALUMINUM END RIBS • G/E SPAR AND NOSE • LEADING EDGE SKIN STABILIZED BY 1-IN.-THICK SLICED HONEYCOMB CORE RIBS	• SPARS, RIBS, SKINS HONEYCOMB SEPARATELY FABRICATED • ALL COMPONENTS ASSEMBLED THROUGH BONDING	• USES PROVEN TECHNOLOGY, LOW MANUFACTURING RISK • ASSEMBLY COST LESS THAN METAL VERSION BECAUSE OF FEWER PARTS • COST APPARENTLY IS 10 PERCENT LESS THAN METAL VERSION • C-SCAN INSPECTION FOR DEFECTS POSSIBLE • REPAIR NOT DIFFICULT ATP MAY 1974 – 31 JANUARY 1975, SUCCESSFULLY PASSED TESTS, "COMPOSITE LOW COST SECONDARY AIRFRAME STRUCTURES," AUGUST 1976, FINAL REPORT AFFDL-TR-76-4, VOLUMES I AND II, AFFDL – WRIGHT PATTERSON AFB, OHIO, CONTRACT NO. F33615-74-C-5111
ADVANCED STRUCTURAL DESIGN FOR FIGHTER COMPOSITE WING BOX F-16 GENERAL DYNAMICS FORT WORTH DIV., TEXAS	• COMPOSITE LOWER SKIN INTEGRAL WITH COMPOSITE SPARS FROM INBOARD PYLON LOCATION TO OUT-BOARD WING TIP • ALUMINUM SPARS INBOARD OF INBOARD PYLON ATTACH TO EMBEDDED TITANIUM INSERT • USE BUFFERED CONSTRUC-TION TO ABSORB DAMAGE	• ASSEMBLY OF LOWER SKIN AND SPARS IN A MOLD • USE AIR BLADDER AND RUBBER EXPANSION TO APPLY PRESSURE DURING CURE IN AN OVEN • UPPER SKIN IS MECHANI-CALLY FASTENED TO THE SUBSTRUCTURE	• INITIAL TOOLING IS COSTLY • LOW MANUFACTURING RISK THROUGH ELIMINATION OF VACUUM BAG AND AUTOCLAVE • COST WILL BE LESS THAN METAL VERSION AS PER LEARNING CURVE • WEIGHT SAVING ABOUT 8 PERCENT • REPAIRABLE BECAUSE UPPER SKIN MAY BE REMOVED ATP JANUARY 1977 – APRIL 1979, QUARTERLY REPORT OCTOBER 1977 – JANUARY 1978. CONTRACT NO. F33615-77-C-3042

TABLE 3-1

SOME ADVANCED COMPOSITES APPLICATIONS IN AIRCRAFT STRUCTURES (CONT)

PROGRAM	STRUCTURAL CONCEPT	MANUFACTURING CONCEPT	COMMENTS
COMPOSITE STRUCTURE OF THE F-16 FORWARD FUSELAGE GENERAL DYNAMICS, FT. WORTH DIVISION, TEXAS FUSELAGE SECTION FROM STA 60 TO STA 227 INCLUDES FORWARD ELECTRONICS BAY, COCKPIT SECTION, EQUIPMENT BAY, AND A FUEL TANK	• THE FUSELAGE IS DIVIDED INTO MANY COMPONENTS • EACH COMPONENT IS SEPARATELY FABRICATED • COMPONENTS ARE THEN MECHANICALLY FASTENED TOGETHER • GR/EP, KEVLAR, AND FIBER-GLASS ARE USED AS MATERIALS. SOME OF THE STIFFENERS ARE OF KEVLAR AND SOME OF GLASS • BULKHEADS ARE OF SAND-WICH CONSTRUCTION	• FEMALE MOLD ON WHICH THE SKINS ARE LAMINATED • RUBBER MANDRELS FOR THE HAT STRINGERS ARE USED. RUBBER MANDRELS ARE POSITIONED OVER LAMINATED SKINS, AND HOT LAMINATED OVER IT • PART IS BAGGED AND AUTOCLAVED	• COST IS ABOUT THE SAME AS ALUMINUM VERSION. GR/EP VERSION MAY RESULT IN ABOUT 10 PERCENT COST SAVING PROVIDED COST OF GR/EP PREPREG TAPE DROPS BY 50 PERCENT • WEIGHT SAVING IS CLAIMED TO BE 20 PERCENT • REPAIRABLE BECAUSE OF ACCESSIBILITY AND REPLACEMENT OF DAMAGED COMPONENTS ATP AUGUST 1973 – NOVEMBER 1976, FINAL REPORT OCTOBER 1977, REPORT NO. AFFDL TR-77-67, CONTRACT NO. F33615-73-C-5130
COMPOSITE HORIZONTAL TAIL OF THE F-16 GENERAL DYNAMICS FT. WORTH DIVISION, TEXAS	• THE HORIZONTAL STABILIZER IS OF THE SINGLE SPAR TYPE • THE SPAR CONSISTS OF A STEEL MEMBER TO WHICH GR/EP ARE LAMINATED • FULL-DEPTH CORE IS USED • SKINS ARE BONDED TO THE CORE	• THE SPAR CONSISTS OF GR/EP COCURED TO A RECTANGULAR STEEL SPAR • SPAR IS FITTED INTO A TOOL AND LOWER SKIN IS LAMINATED, BONDED, AND COCURED TO SPAR • THE LOWER SKIN IS INVERTED AND A FULL-DEPTH NOMEX HONEY-COMB CORE IS PREFITTED AND BONDED TO CORE • UPPER SKIN IS LAMINATED OVER CORE AND SPAR WITH AN ADHESIVE FILM IN BETWEEN AND IS COCURED IN PLACE	• DESIGN IS SIMPLE • COST MAY BE LOWER • LIGHTER WEIGHT • REPAIRABLE BECAUSE OF SIMPLICITY OF STRUCTURE ATP AUGUST 1973 – NOVEMBER 1976, FINAL REPORT OCTOBER 1977 REPORT NO. AFFDL TR-77-67, CONTRACT NO. F33615-73-C-5130

development composite wing and proof of structural integrity must be demonstrated by full-scale testing.

Durability

For the purpose of the technology assessment, durability can be defined as the capability of structure to maintain its structural integrity throughout its intended service life with reasonable maintenance costs.

The durability of metal structures is usually measured in terms of fatigue strength and resistance to pitting, stress, intergranular, and other forms of corrosion. Composites exhibit different modes of damage such as delamination, matrix crazing/cracking, fiber failure, void growth, fiber/matrix disbonding, and composite cracking.

Data on the durability of aluminum alloys have been accumulated for about 50 years. The capability to design efficient fatigue-resistant structures by means of establishing the proper working stress levels and avoiding high stress concentrations has progressed to an advanced state. Intergranular and stress corrosion problems have been minimized by new tempering processes which provide virtual immunization against these types of corrosion, while improvements in surface coatings have drastically reduced costs associated with pitting corrosion.

As a minimum, the airlines expect equal durability performance from composite materials. Although graphite/epoxy composite materials have been demonstrated to have excellent resistance to the civil air transport environment, long-term exposure data in this environment have not yet been accumulated. The fatigue strength of graphite/epoxy composite materials is superior to metal structure for cyclic loads in the plane of the laminate, but their fatigue strengths for the interlaminar shear and flatwise tension failure modes are significantly inferior to the in-plane composite cracking mode. At present, the airlines and the manufacturers want to see far more data to prove the durability of graphite/epoxy composite wing box structure, particularly since wing components are permanently joined to the fuselage structure and cannot be economically replaced.

The FAA position, as stated in Reference 1, is that structural integrity safety levels shall be at least as high as for metal structures. The retention of material strength after long-term environmental exposure must be proven by sufficient tests on components, subcomponents or coupons to establish the fatigue scatter and environmental effects. Full-scale fatigue tests, accounting for effects of the appropriate environment, must be conducted to substantiate the fatigue strength.

Five cases have been identified where the technology does not exist to properly measure the durability of graphite/epoxy composite structures:

1. The vulnerability to impact damage
2. The resistance to loads normal to the plane of the laminate
3. The long-term effects of environment on material allowable strengths
4. The effect of cyclic loads on component stiffness
5. The effects of imperfections resulting from material and manufacturing deviations.

Impact damage is a key issue because of the constant exposure of the wing structure to runway debris, tire/wheel fragments, hailstones, and other foreign objects as well as to damage inflicted by machinery or personnel while servicing or maintaining the aircraft.

An assessment of the durability of a composite wing for a particular aircraft with respect to impact damage entails an investigation of several phenomena:

- A determination of the impact frequencies for a range of impact energies and forms as a function of location on the wing box. Experimental evidence is desirable and can probably be obtained only by some type of in-service flight evaluation.
- The degree of damage sustained by the composite structure as a function of the impact forces. Methodology and supporting experimental data should be developed to predict the amount of damage.

- The residual strength of the impacted structure. This phenomenon will be further discussed in the damage tolerance assessment.

Graphite/epoxy composite materials do not exhibit superior durability for the interlaminar shear (Reference 3) and flatwise tension failures modes. A durable design can be created by avoiding internal loads in the structure which produce significant interlaminar forces. However, wing structures contain some rather complex design features, and the flexing of the wing under load can often induce internal loads within the structure which are normal to the plane of the laminate and are not readily identified by the current analysis methods. If these loads are not accounted for in the design, delamination may occur, which could cause a joint to fail or reduce panel stability. Interlaminar shear forces can usually be controlled by the ply pattern and stacking sequences which unfortunately may also increase manufacturing costs. Data on interlaminar shear fatigue strength will be necessary for tradeoff studies of durability with manufacturing costs.

The long-term effects of environment on the allowable strength of composite material can only be proven with the passage of time. Current NASA ACEE and other composite technology programs should address this problem. More on this subject is discussed as a materials issue.

Some test data have indicated the possibility that the stiffness of graphite/epoxy composite structures may be noticeably reduced after the structure is exposed to a high number of repeated load cycles.

Army helicopter rotors measured before and after cyclic loading tests have shown a significant loss in stiffness. The stiffness of the DC-10 upper aft composite rudder was measured before test and again after exposure to 108 random vibrations of 350-to 1800-Hz bandwidths. A reduction in stiffness of approximately 10 percent was attributed to a disbond of shims at the hinge bracket installation, but could have been partly due to reduced material stiffness properties of the composite laminate. Such changes in bending and torsional stiffness after long-term exposure to a service load environment could result in unsafe flutter speed margins unless properly accounted for in the design.

Current NASA ACEE programs on medium primary structure must investigate the effects of cyclic loads on structural stiffness and perform tests to determine if changes in stiffness occur in order to substantiate flutter speed margins and provide for safe aircraft operation. Much of the existing technology gap should be resolved by these programs, but they should be supplemented with test specimens representative of wing structure.

Defects can also be introduced in the composite structure during the manufacturing process which can degrade the durability of the structure. Data should be developed to establish durability as a function of the product quality level. Other data should be developed to establish the relationship between quality and manufacturing cost. Together, the data will permit a tradeoff between structural weight and manufacturing costs since for a given durability criterion, the design strain levels are influenced by the structural quality of the product. Additionally, the data will assist in the engineering disposition of manufacturing deviations, as discussed later.

In summary, much of the durability technology gap for composite wing structures will need to be resolved by other composite technology programs, particularly the question of strength degradation due to long-term environmental exposure and the reduction of stiffness due to cyclic loads. Some contribution will be made to interlaminar shear fatigue strength and the effects of defects on composite wing durability. Plans must be made to resolve the impact damage issue.

The composite wing technology program will require provisions for acquiring durability test data, exercising the capability for designing durable structure, and demonstrating durable qualities by means of a full-scale fatigue test and an in-service flight evaluation.

Damage Tolerance

The damage tolerance of a structure is a measure of its capability to sustain loads in the presence of damage. For civil transport aircraft, the level of damage tolerance required by the FAA is specified in Reference 4. In Reference 2, the FAA emphasizes the importance of experience with previous damage tolerance designs, constructions, test, and service usage. To date, this experience has not been accumulated for civil transport committee primary wing structure.

For purposes of this study, the damage tolerance capability of graphite/epoxy laminates is assessed for the damage modes discussed in the durability assessment: delamination, matrix crazing/cracking, fiber failure, void growth, fiber/matrix disbond, and composite cracking. Fatigue, environmental exposure, and accidental damage must be considered.

The extent of permissible damage must be consistent with damage detection capability. The development of damage detection methods using nondestructive testing technology will be discussed later. In areas where the damage is not easily detectable or measurable, the structure must be designed to ensure an extremely low probability of failure throughout its operational life. The damage tolerance assessment is based upon the assumption of the initial presence of flaws or defects associated with a design quality level.

The main body of research into the damage tolerance of graphite/epoxy structure appears to have been conducted by investigators with a previous background in the field of fracture mechanics as applied to metal structures. Much of their work deals with the residual strength of through-cracks, holes, and cutouts subjected to both tension and compression loads. Crack growth does not appear to be a problem since available data (Reference 4) showed no crack growth at the tip of an induced sharp stress concentration after 10^6 cycles at 80 percent of static ultimate load for a [0, 90] pattern. Therefore, the presence of through-cracks in the structure should be from a discrete damage source rather than from the gradual crack propagation often experienced with metal structures.

A damage tolerance study was made for the lower wing panel of the conceptual wing described in Section 6 using linear elastic fracture mechanics methods to demonstrate the existing methodology. The study assumed a through-crack which severed both the cover skin and the integral blade stiffener being subjected to tension loads. In order to apply the linear elastic fracture mechanics methods, the material was modeled as a homogeneous, linear elastic, anisotropic continuum exhibiting a simple enlargement of the crack without branching or directional change. The study results shown in Figure 3-3 indicate that design features such as low-modulus crack arrestment strips are probably necessary for the conceptual design to satisfy FAA damage tolerance criteria.

The residual strength of graphite/epoxy panels subjected to impact damage has also been extensively investigated. The investigations cover several impact forms and various energy levels for different layup patterns and thicknesses. The improvement in impact resistance by the addition of layers of Kevlar 49 fabrics has also been established. Methodology has been developed to predict the residual tension strength of graphite/epoxy laminates subjected to impact damage from a known form and energy source. The residual compression strength cannot be similarly predicted (References 5 and 6).

There is little information available on another major area of concern in damage tolerance. This deals with the reduction in compressive residual strength through instability due to panel delamination, fiber disbond, or the loss of an integrally cured joint which connects a panel to a stabilizing member. Applied to induced loads which produce interlaminar shear or flatwise tension can promote initiation or growth of these types of failures and progressively reduce the residual instability strength of the structure to less than the strength required for design limit load. Much more emphasis needs to be placed on research directed toward the determination of the growth and compressive residual strength characteristics of structure with delaminations which are initiated by impact damage, or interlaminar shear, or flatwise tension forces.

Much of the methodology and experimental test data to support the design and verification of damage-tolerant composite structures must be developed in order to qualify the current NASA ACEE medium primary structural program for FAA certification. Additionally, the AV-8 Harrier wing (Reference 7), the F18A wing, the B-1 horizontal stabilizer, and other military aircraft composite structure require extensive development to provide compliance with similar military damage-tolerance requirements.

The composite wing structure of a civil transport aircraft will be sufficiently different from these earlier programs to require further development of

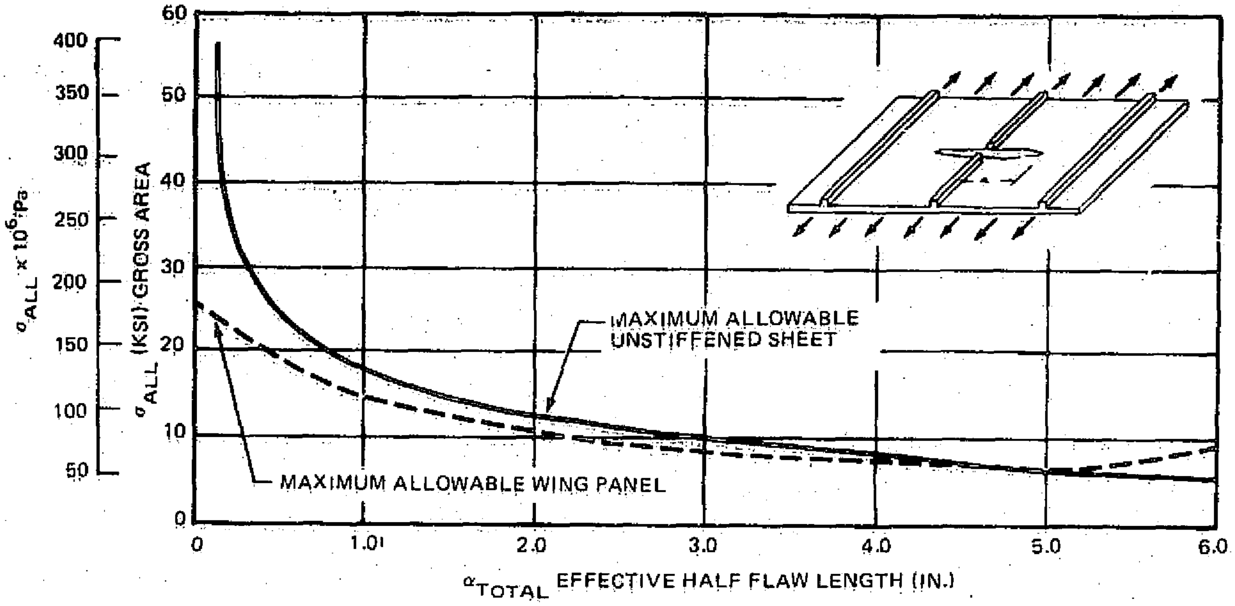

FIGURE 3-3. COMPOSITE WING RESIDUAL STRENGTH

methodology correlated with experimental test data in order to develop an efficient wing design with adequate damage-tolerance capability. The structural integrity can only be proven by full-scale damage-tolerance tests during a wing technology development program to demonstrate this capability to the satisfaction of the manufacturer and the airlines before embarking on a production program.

Crashworthiness

The FAA criterion for crashworthiness of the airframe is to ensure that occupants have every reasonable chance of escaping serious injury under realistic and survivable crash conditions (Reference 3). Airframe structures which utilize advanced composite material must provide the same level of safety as conventional construction (Reference 2).

For wing structures, the goal is to avoid fuel spillage from the integral wing tanks by a design that maintains fuel tank integrity for a reasonable set of crash conditions or off-runway incidents. The following failure conditions must be considered:

1. The tank within the fuselage contour must be protected so that exposure to ground scraping action is unlikely for a wheels-up landing,

2. The tank within the fuselage contour must be capable of sustaining 9-g forward crash loads,

3. Airframe components supported by the main wing box integral tank structure must be designed to break away from the wing box without rupturing the wing tank.

This crashworthiness criterion is satisfied for a DC-9 aircraft with advanced composite wing box structure in our current NASA ACEE study.

The DC-9 center wing box is located inside the fuselage and has a fuel capacity of 3.528 cubic meters (932 gallons). The tanks are protected against scraping during a wheels-up landing by the fuselage shell and the heavy main keel member in the wheel well. The lower surface of the wing is 58.42 centimeters (23 inches) above the lower fuselage loft line at the front spar and 43.18 centimeters (17 inches) above the rear spar. In addition, there are two cant beams plus the keel beam directly under the wing which shield the wing box structure (see Figure 3-4).

In over 15 million flight-hours accumulated by the DC-9 fleet, there has been no damage to the center wing box in survivable incidents. The composite wing box is afforded the same level of protection as the conventional wing box and the lower ductility of composite materials is not a factor.

Inertia fuel crash loads of 9g must be sustained by the wing box structure inside the fuselage. Pressure loads are derived based on a full fuel tank with a 9-g head for this condition. The less ductile characteristics of composites can be accounted for in the detail design, however, and do not impose any special design problems.

Fusing of conventional structures allows specific components to break free at predetermined load values to preclude penetration or other damage to the fuel tank, landing gear support fittings, and some flight control fittings.

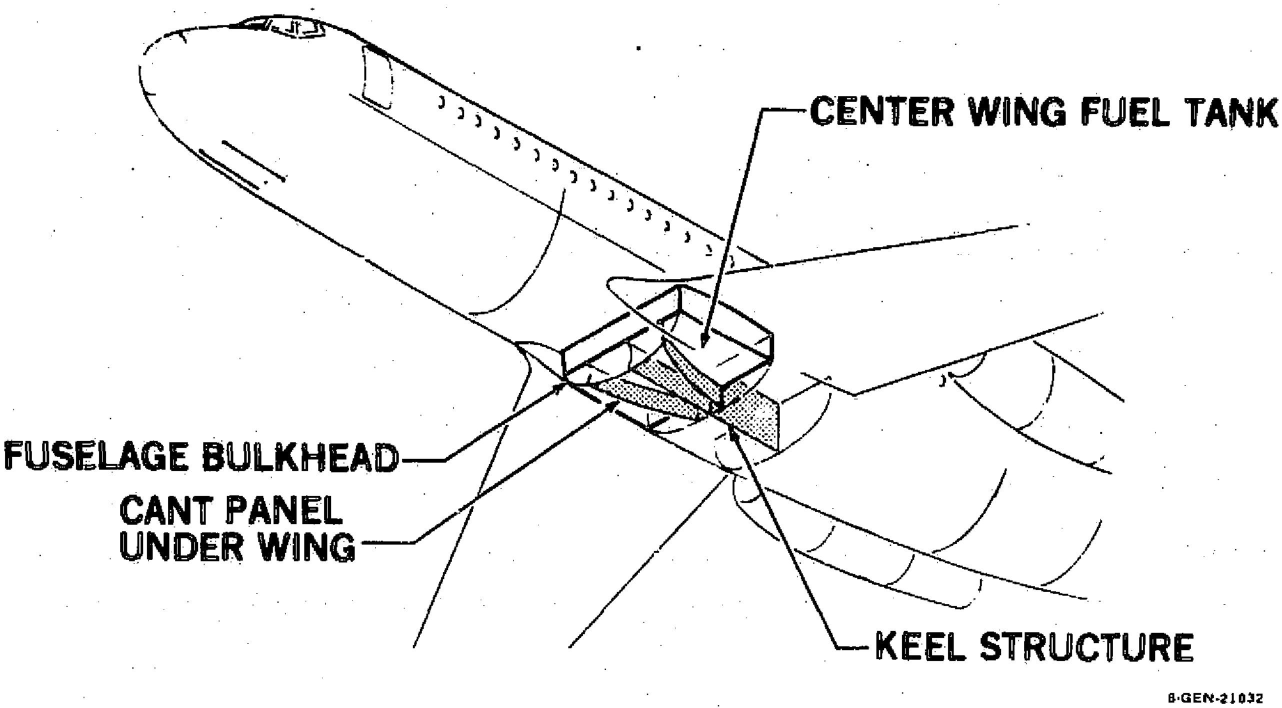

FIGURE 3-4. PROTECTION OF CENTER WING FUEL TANK STRUCTURE – DC-9 AIRCRAFT

The landing gear may be subjected to loads far in excess of design loads after contact with ditches, runway light standards, or other obstacles when involved in off-runway incidents. These incidents are infrequent but must be accounted for in the design to prevent fuel tank rupture in accordance with FAR 25.721. The DC-9 main gear is designed to fail in the gear cylinder for high-drag load conditions, but other fuse points must be utilized for high resultant vertical and drag load combinations. The concept used for the DC-9 composite wing design allows for the main gear to remain intact. The failure will occur aft of the tank boundary in the following sequence:

- The lower cover skin and titanium doubler will fail aft of the rear spar.
- The two tension bolts attaching the support fitting to the lower bulkhead cap will fail.

- The shear bolts attaching the support to the rear spar web will fail in the thin bolt heads.

- The upper two bolts and upper cover skin and doubler will bend upwards as the intact gear and support fitting rotate upwards due to lack of a restraining moment.

A conceptual breakaway design for a composite wing box structure is illustrated in Figure 3-5. The primary tank is designed to ensure that the bulkhead inside the tank is stronger than the two tension bolts attaching the supporting fitting to the bulkhead.

The wing flaps in the landing flap position and wing-mounted engines will contact the runway if the landing gear collapses during landing. FAR 25.963 specifies that fuel tank integrity must be maintained for this condition.

The DC-9 wing flap is attached to the main composite wing box at three support locations. Four bolts attach the hinge fittings at each location (see Figure 3-6). The lower two bolts at each hinge fitting are necked down

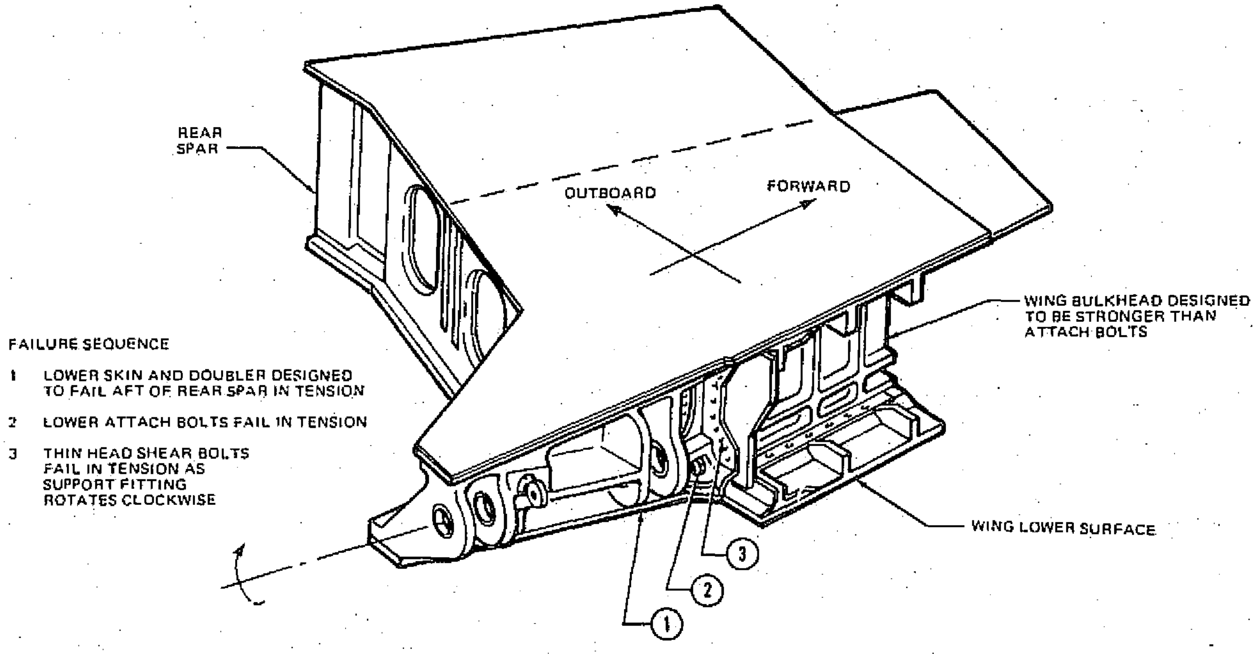

FIGURE 3-5. DC-9 MAIN LANDING GEAR SUPPORT STRUCTURE OVERLOAD BREAKAWAY DESIGN

outside the tank boundary to form a fuse point for high-tension loads caused when the flap structure strikes the runway. The fitting will then rotate about the two upper bolts and the wing tank will not rupture. The primary design task is to ensure that the flap bulkhead inside the tank is stronger than the fuse point of the attach bolt.

Crashworthiness design of composite wing structure appears within the state of the art, but should be exercised and demonstrated as part of a composite wing technology program.

Repair of Major Damage

To date, all composite structural components built and installed on flight aircraft have been designed so that in the event of major damage to a component, it could be readily removed from the aircraft and replaced with a new part. The damaged part could either be scrapped or repaired. Autoclave facilities could be used if necessary. For this reason, repair technology has been developed only for lesser damage which can be economically repaired on the aircraft.

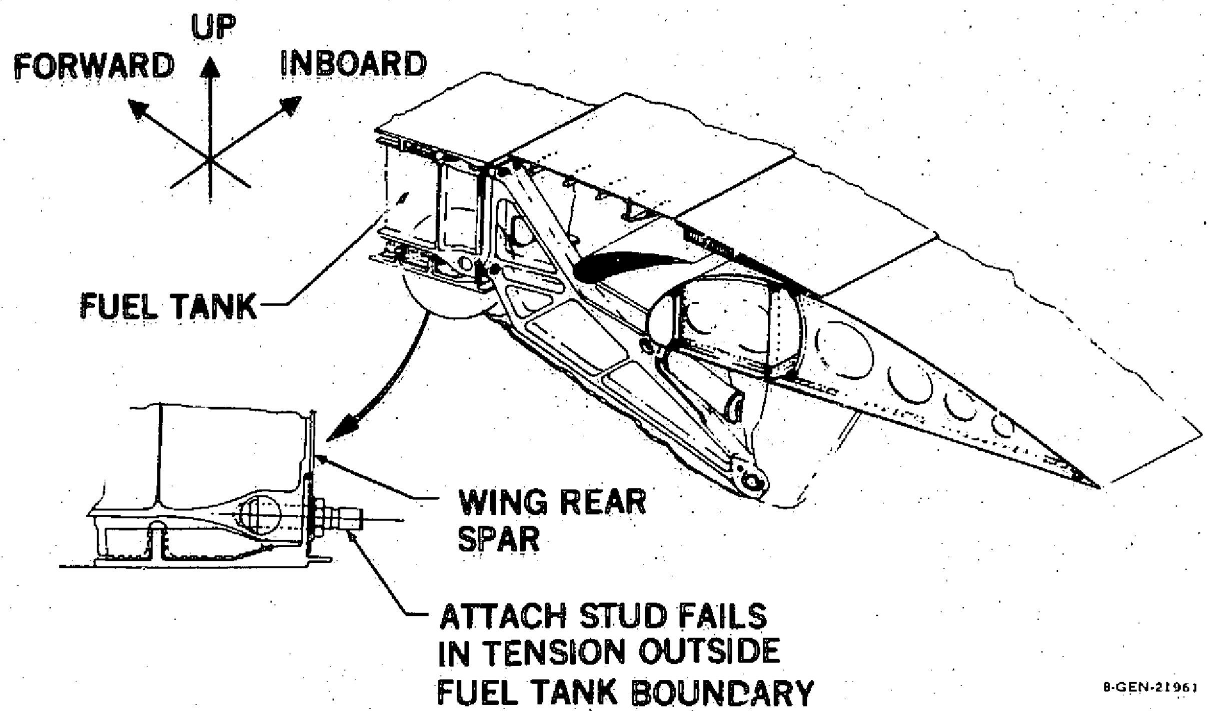

FIGURE 3-6. DC-9 FLAP SUPPORT STRUCTURE OVERLOAD BREAKAWAY DESIGN

A survey of current programs investigating damage repair technology for military and commercial application is presented in Table 3-2.

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TABLE 3-2
REPAIR TECHNOLOGY PROGRAM SURVEY GRAPHITE/EPOXY AIRCRAFT STRUCTURES

COMPANY	AGENCY	TITLE AND CONTRACT NO.	SUMMARY AND REPORT NO.
NORTHROP CORPORATION HAWTHORNE, CA	AIR FORCE FLIGHT DYNAMICS LAB WRIGHT-PATTERSON AFB, OHIO	LARGE AREA COMPOSITE STRUCTURE REPAIR F33615-76-C-3017	LARGE-AREA DAMAGE IS DEFINED TO BE HOLES 4 INCHES TO 12 INCHES IN SIZE. DATA PRESENTED ON PROPERTIES OF LAMINATES AND ADHESIVES. TEST RESULTS OF VACUUM BAG CURED SCARF JOINTS WITH AND WITHOUT DOUBLERS ARE PRESENTED. EFFECT OF MOISTURE AND TEMP IN REDUCING THE JOINT STRENGTH IS COVERED. REPORT NO. AFFDL-TR-77-5, MAY 1977, JULY 1977 ATP JUNE 1976, R&D STILL IN PROGRESS
LOS ANGELES AIRCRAFT DIV., ROCKWELL INTERNATIONAL CORPORATION	AIR FORCE MATERIALS LAB WRIGHT-PATTERSON AFB, OHIO	ADVANCED COMPOSITES DESIGN GUIDE VOL. III-MFG. JAN. 1973 F33615-71-C-1362	A COMPREHENSIVE STUDY OF REPAIR TO AIRCRAFT STRUCTURE IS PRESENTED. THIS REPORT APPLIED TO BORON/EPOXY. HOWEVER, SOME BASIC DESIGN CONCEPTS, INSPECTION METHODS, ETC., ARE APPLICABLE TO ALL TYPES OF COMPOSITE. ONE REFERENCE ONLY IS GIVEN, REPAIR TECHNOLOGY FOR BORON/EPOXY COMPOSITES. REPORT NO. AFML-TR-71-270, CONTRACT NO. F33615-69-C-1498, GRUMMAN AEROSPACE CORPORATION
DOUGLAS AIRCRAFT COMPANY, DIV. OF MCDONNELL DOUGLAS CORPORATION LONG BEACH, CA	NASA WASHINGTON DC	BOLTED JOINT IN G/E COMPOSITES NAS1-13172	SPECIMENS WITH OPEN HOLES WERE TESTED IN TENSION TO DETERMINE STRESS CONCENTRATION INDUCED BY THE PRESENCE OF THE HOLE. DIFFERENT BOLTED JOINT DESIGNS WERE TESTED. ANALYTICAL STUDIES OF THE RESULTS WERE MADE AND REDUCED FOR DESIGN APPLICATION. REPORT NO. NASA CR-144899, JAN 1977
BOEING COMMERCIAL AIRPLANE COMPANY SEATTLE	NASA WASHINGTON, DC	DESIGN FAB OF G/E BOLTED WING SKIN SPLICE SPECIMENS NAS1-14327	THE PROGRAM WAS CONDUCTED TO DELIVER SEVEN GRAPHITE/EPOXY BOLTED JOINT SPECIMENS OF DIFFERENT CONFIGURATIONS TO NASA-LANGLEY FOR TEST PURPOSES. SOME OF THE JOINT DESIGN COULD BE USED IN REPAIRS. REPORT NO. NASA CR-145216, MAY 1977
VOUGHT CORPORATION SYSTEMS DIV. DALLAS	AIR VEHICLE TECHNOLOGY DEVELOPMENT NADC WARMINSTER, PA	S-3A G/E SPOILER FAB OF TEN SHIPSETS AND DAMAGE REPAIR STUDY N62269-75-C-0428	DIFFERENT CASES OF DAMAGE AND WEAR OF SANDWICH CONSTRUCTION ARE CONSIDERED. THE SPOILER STRUCTURE IS DIVIDED INTO ZONES, AND THE EXTENT OF DAMAGE REQUIRING REPAIR IN EACH ZONE IS DIFFERENT. SKIN IS PATCHED WITH PREPREG GRAPHITE/EPOXY AND FG/E. VACUUM BAG AND HEATING BLANKETS ARE USED FOR CURING THE RESIN. CORE DAMAGE IS REPAIRED BY A CLEANUP FOLLOWED BY POTTING. A COOKBOOK APPROACH TO REPAIRING IS ALSO PRESENTED. REPORT NO. NADS-76234-30, MAY 1976, FINAL TECH REPORT, ATP 74 – FLYING ON S-3A AIRCRAFT
GENERAL DYNAMICS FOR WORTH DIV. TEXAS	AIR FORCE FLIGHT DYNAMICS LAB WRIGHT-PATTERSON AFC, OHIO	REPAIR PROCEDURES FOR ADVANCED COMPOSITE STRUCTURES F33615-74-C-5133	DEVELOPMENT WORK IN PROCEDURES FOR REPAIRING STIFFENED AIRCRAFT STRUCTURES IS DISCUSSED. FABRICATION OF STRUCTURAL COMPONENTS FOLLOWED BY DAMAGING AND THEN REPAIRING AND TESTING ARE REPORTED. REPAIR OF HYBRID LAMINATE IS ALSO COVERED. DAMAGE AREA IS ON THE ORDER OF 3-INCH SIZE HOLES. REPORT NO. AFFDL-TR-76-57, DEC 1976, FINAL REPORT IN TWO VOLUMES ATP JUNE 1974 FEB 1976.
MCDONNELL AIRCRAFT COMPANY DIV OF MCDONNELL DOUGLAS CORPORATION ST LOUIS		COMPOSITE DEVELOPMENT PROGRAM G/E REPAIR CONCEPTS, VOL IV IRAD	REPAIRS IN THIN AND THICK LAMINATES WERE MADE AND TESTED. BOTH BRITTLE AND DUCTILE ADHESIVES WERE TESTED. TWO TYPES OF PATCH WERE USED, ONE USING TITANIUM FOIL AND THE OTHER GRAPHITE/EPOXY. THE THICK LAMINATE WITH A 4-INCH-DIAMETER HOLE WAS REPAIRED USING A LAMINATION OF TITANIUM PLIES AND ADHESIVE. IN ADDITION TO THE BONDING, EACH PLY WAS SEPARATELY BOLTED TO THE PARENT LAMINATE TO REDUCE PEELING ACTION, AS WELL AS PICK UP SOME SHEAR. STUDY OF MATERIALS WITH EXTENDED SHELF LIFE WAS ALSO MADE. REPORT NO. MDC A3715-27, MARCH 1976
DOUGLAS AIRCRAFT COMPANY, DIV OF MCDONNELL DOUGLAS CORPORATION LONG BEACH, CA		REPAIR TECHNIQUES FOR G/E COMPOSITES IRAD	THE REPORT IS A COMPILATION OF AVAILABLE INFORMATION WITH RESPECT TO REPAIRING COMPOSITE STRUCTURE LAMINATES AS WELL AS SANDWICHES. DAMAGES ARE ON THE ORDER OF 3-INCH DIAMETER AND UNDER. INSPECTION PROCEDURES AND TEST RESULTS ARE ALSO GIVEN. REPORT NO. MDC J7246, SEPT 1977
BELL HELICOPTER FORT WORTH, TX	AFML	DURABILITY OF ADHESIVE BONDED JOINTS AF33615-71-C-1168	ADHESIVES FOR BONDING METAL-TO-METAL AND COMPOSITE-TO-COMPOSITE WERE EVALUATED AND PROCESSES DEVELOPED. INFORMATION IS APPLICABLE TO REPAIR OF COMPOSITES. JUNE 71 TO MARCH 74, REPORT NO AFML-TR-74-26
BOEING COMMERCIAL AIRPLANE COMPANY, SEATTLE	NASA LANGLEY RESEARCH CENTER, HAMPTON VA	ADVANCED COMPOSITE STABILIZER FOR BOEING 737 AIRCRAFT NAS1-15025	IT IS INTENDED TO FABRICATE STIFFENED PANELS, WHICH WILL THEN BE DAMAGED, REPAIRED, AND TESTED, TO DEMONSTRATE REPAIRABILITY OF STRUCTURE. JULY 1977, R&D STILL IN PROGRESS. FIRST QUARTERLY TECHNICAL PROGRESS REPORT, 18 OCT 1977. (18 JULY 1977 THROUGH 18 OCT 1977)

A composite wing box for a commercial transport aircraft is built into the fuselage structure in a manner which makes wing replacement extremely costly and replacement cannot therefore be considered as a viable alternative to repair, nor can a throwaway aircraft be considered. As stated earlier, one of the clearly defined conditions for airline acceptance is that the composite wing box structure can be repaired and returned to service. Repair costs and downtime should compare favorably with those of conventional wing structure. For extensive damage, temporary field repairs can be made for a ferry flight to a major repair depot.

For study purposes, the assumption is made that major damage will occur in a region where the damaged structure can be cut out of the airplane as required and repairs effected with new structure and/or repair doublers which are either bolted or bonded together.

Repair criteria are established which exclude attachment of primary wing members by adhesive bonding only. Primary wing structure repair joints must have mechanical fastener strength to sustain limit loads. Adhesive bonding of the joint may then be utilized to develop strength to sustain ultimate design loads. The adhesive bonds used for this purpose must have a demonstrated long life. Based on current bonding technology, this implies that the bonds must be made under heat and pressure, since present cold-bond systems have not shown the required durability in a service environment.

Specimen tests have shown that the strength of composite structure decreases as bearing stresses increase. Ultimate strain design levels for basic wing structure will probably be established at around 4000 μin./in. to allow for damage from foreign objects or small unloaded holes. Lower design strain levels would proportionately reduce weight savings. Bolted-on repair doublers would produce high bearing loads in the parent structure which would reduce allowable strain levels below those required to sustain ultimate design loads.

One approach to resolve the problem is to bond doublers to the parent structural member adhesively to reduce strain levels. After the stress level is reduced in the parent structure, bolts can be installed to provide strength for design limit load.

Although the solution can be simply stated, it remains to be accomplished. Technology development is required to select adhesives, to determine heat and pressure requirements for bonding, field cutting, and drilling tools, and nondestructive inspection methods. Design studies must be conducted to establish the sizes of typical repair members and accessibility requirements. Joint load distribution must be investigated to ensure joint strength integrity and durability. Much of this technology must be available during the design synthesis phase to establish a repairable structural arrangement.

Conflicting data have been obtained on the capability of interference fit fasteners to increase the ultimate strength and durability of joints. Preliminary results of tests being currently conducted at NASA-Langley on composite joints attached with 0.48-cm (3/16-inch) taper-lok fasteners indicate the possibility that interference fit fasteners can provide better repair strength than clearance fit bolts. Further investigations should be undertaken in this area as soon as possible.

Considerable testing will be required to provide repair technology data and to prove to the airlines that proposed repair methods are viable and compare favorably with methods now used to repair conventional wing structure. Tests will be selected to supplement repair data being made available from other programs.

Three areas for test investigation are recommended:

1. Specimen tests of joints utilizing various candidate interference fit fasteners. Test variables will include the amount of interference fit, specimen thickness, environmental effects, type of loading, and with and without adhesive bonds.

2. Subcomponent panel tests of bolted and bonded joints subjected to combined loadings (e.g., compression and shear) representative of typical wing load conditions.

Panels would be designed to represent typical composite wing box construction and would be fabricated by methods to be utilized to repair composite wing damage at major repair depots. Autoclave methods should not be used.

3. The major subcomponent test article will be repaired after the crashworthiness test failure and retested to verify static strength capabilities. The full-scale fatigue and damage tolerance test article may also be repaired and retested.

It is believed that this program will provide the data base required for the design synthesis of a composite wing box design for a new production aircraft, provide evidence for airline acceptance, and satisfy FAA compliance requirements for type certification of the development plan flight article.

Thermal Incompatibility

One of the material characteristics of graphite fibers is that they exhibit a decreasing coefficient of thermal expansion with an increasing fiber modulus. The high-modulus fibers actually have a negative coefficient of thermal expansion and provide a quasi-isotropic laminate with a near-zero thermal expansion.

A problem related to thermal cycling of a composite wing is that the large difference in expansion coefficients between graphite/epoxy composites and metallic alloys (aluminum, titanium, etc.) induces internal structural loads at interfaces where the structures are bonded or mechanically attached.

After components of dissimilar materials such as metallic and graphite/epoxy composites are joined by a number of rows of fasteners or by adhesive bonding, thermal loads develop in the components (and the attachment medium) with excursions from ambient temperature.

Static and fatigue analyses which account for internal thermal loads in conjunction with external load conditions must be performed to assure structural integrity. This can be particularly significant at the end fasteners in a hybrid joint. This location is fatigue-critical for externally applied loads and is also the location where the highest bolt loads occur due to thermal loads. A computer analysis is usually required to determine the redundant bolt load distributions. The stiffness of the structure, hole clearances, and friction all affect the final load distribution in a bolt pattern. This type of analysis is not normally required for commercial transport, and design and analytical methods must be updated to include this capability. Tests will need to be conducted to verify the analytical predictions.

The need for thermal load analyses will decrease with the utilization of composite structure. Whereas early components may feature a composite wing box structure in combination with metallic leading and trailing edge components, later models will probably convert the secondary structure to the same advanced composite materials.

Wing structural interfaces to be examined for the effects of thermal loads include the following: wing leading edge, slat, flap, main landing gear support fitting, spoilers, ailerons, wing trailing edge, and fuselage. The design must include flexibility to avoid the buildup of thermal loads and provide structural capability to resist the loads.

The thermal incompatibility technology issue for composite wing box structures will be largely resolved by current NASA ACEE composite programs which feature hybrid structure. In addition, the technology will be further exercised by the accomplishment of the composite wing box development plan, as described in Section 7.

Manufacturing Deviations

Deviations from engineering drawing requirements are inevitable in the manufacturing process. These deviations must be investigated by the design engineering department and one of the following engineering dispositions must be prescribed:

1. Parts may be rejected as unacceptable for strength, fit and function, or quality.

2. Parts may be found acceptable for use even though not in agreement with the engineering drawing.

3. Parts may be reworked to meet engineering requirements.

Common sources of manufacturing deviation for composite structures include the following:

- Voids, delaminations, foreign material inclusions
- Poor resin content
- Improper cure cycle
- Loss of pressure during the cure cycle
- Geometric distortion due to warping and bowing
- Cutting, drilling, and trimming operations
- Surface finish.

Prompt action must be taken by Engineering to support manufacturing operations in order to avoid delays and to control the cost of the final product by minimizing rejection rates, providing low-cost rework instruction, and avoiding the high cost of unscheduled out-of-sequence assembly operations.

The problem of warping and bowing of long composite parts during fabrication must be investigated. Limits of acceptability must be determined with respect to fit and function and structural integrity. Metal parts are frequently straightened by hot-forming to obtain a fit with acceptable residual stress levels. This rework method obviously cannot be applied to the brittle composite materials. Experience in state-of-the-art composites is that they must fit the assembly without undue preload or they must be rejected, undue preload being an undefined quantity for individual parts.

Another related concern is warpage of the completed wing box after it is removed from an assembly jig or molding tool. It is important to prove that the wing box warpage can be controlled or reworked to restore the box to limits which are acceptable for aerodynamic qualities.

Some experience with the disposition of manufacturing deviation will be obtained in ongoing NASA ACEE programs and in-house development work. However, many of the deviations will be unique for a composite wing box structure, and experience with full-scale composite wing box hardware is necessary if the issue is to be resolved.

The composite wing box structural development plan described in Section 7 should identify the potential sources of deviations and provide a realistic demonstration that manufacturing deviations can be disposed of by methods acceptable to Engineering, with a minimum impact on cost and schedule.

Lightning Protection

When the lightning attaches to the aircraft, the structure becomes part of the lightning channel. Since graphite/epoxy structures are 1000 times less conductive than aluminum structures, there is concern that this phenomenon may produce structural damage and aircraft system disturbances in graphite/epoxy structures which are not encountered with conventional aluminum structures.

Aircraft skin panels are subject to static charge buildup. For highly conductive aluminum panels, the charge is quickly dissipated. Less conductive graphite/epoxy panels may slow down the redistribution process and cause precipitation static (p-static) or in-tank arcing problems.

Lightning Strike - When struck by lightning, an aircraft becomes involved in various phases of lightning current transfer. Figure 3-7 shows the critical lightning current waveforms associated with various phases of a severe lightning strike (Reference 8). For the purpose of defining lightning protection requirements, the aircraft surfaces can be divided into three lightning strike zones (Reference 9):

Zone 1: Surfaces of aircraft for which there is a high probability of a direct lightning strike.

Zone 2: Surfaces of aircraft for which there is a high probability of lightning strike being swept rearward from a Zone 1 point of direct strike.

Zone 3: The aircraft areas other than those covered by Zone 1 and Zone 2.

Figure 3-8 shows the lightning strike zones of the DC-9 wing. These zonal regions were defined by an analysis of the natural lightning strike phenomena and laboratory lightning attach point studies using scale aircraft models.

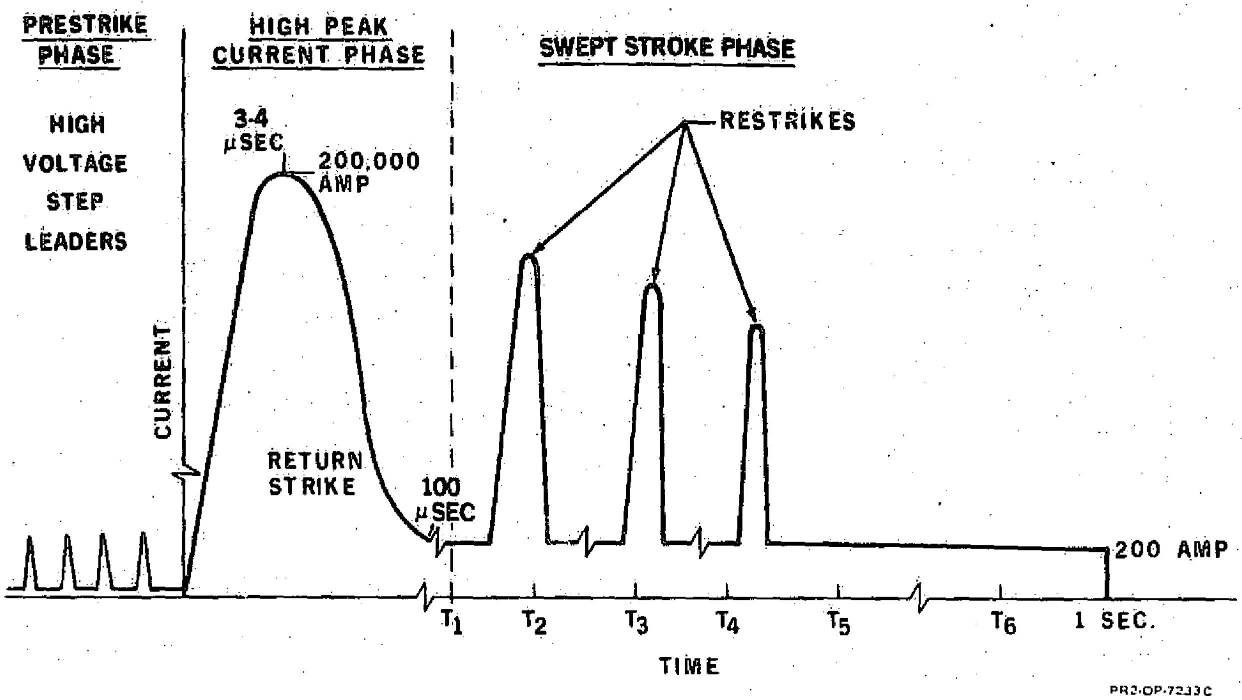


FIGURE 3-7. CRITICAL LIGHTNING STRIKE CURRENT WAVEFORM

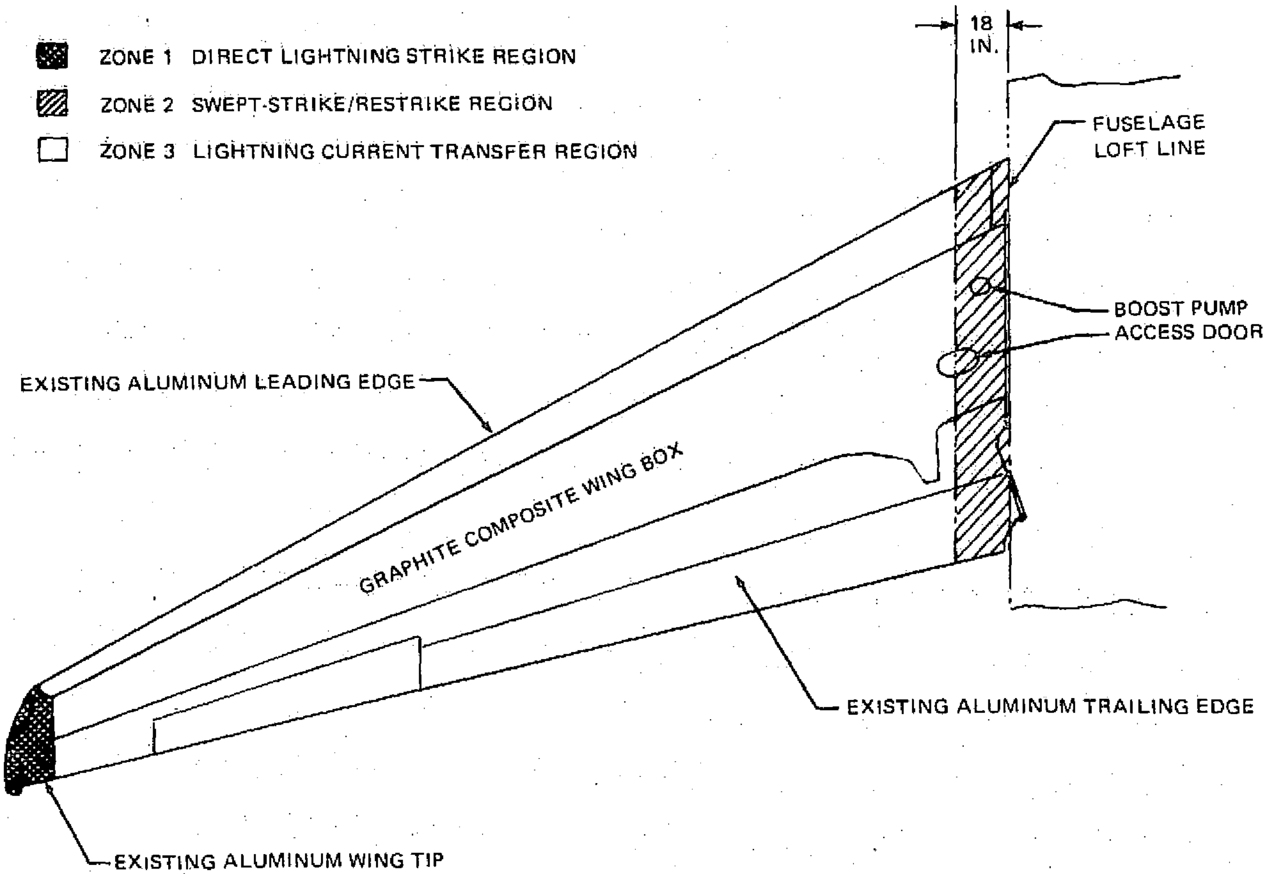


FIGURE 3-8. DC-9 WING LIGHTNING STRIKE ZONES

The wing tip section is located in the Zone 1 direct lightning strike region. The trailing edge of the wing tip section is vulnerable to all three phases of the lightning current waveform shown in Figure 3-7.

The current flow activity of a severe lightning strike can last for 1 second, and during this time, an aircraft traveling at a speed of 805 kilometers per hour (500 miles per hour) could move forward 104 meters (340 feet) in relation to a stationary lightning channel. For a lightning channel initially attaching to the aircraft nose area, the channel would be swept over the fuselage and wing root area, making these surface areas vulnerable to the swept-stroke phase of lightning current waveform shown in Figure 3-7. The wing root area within 46 centimeters (18 inches) of the fuselage loft line is thus classified as the Zone 2 swept-stroke/restrike region (Reference 9).

When a lightning channel attaches to a wing tip, the associated lightning current flows from the wing tip along the wing box structure to other aircraft extremities. Thus, the entire wing box structure is classified as the Zone 3 lightning current transfer region. The electrical and fuel instrument wiring inside the wing structure is also vulnerable to lightning current-induced transient effects.

The existing DC-9 aluminum wing design has incorporated adequate lightning protection designs for the direct lightning strike, swept-stroke/restrike, lightning current transfer, and the lightning-induced transient effects. The user of an all-graphite composite box structure integrated with the existing aluminum wing tip, leading edge, and trailing edge sections will introduce new lightning hazards, and additional lightning protection design considerations must be addressed to ensure the safety of the wing structure and the associated subsystems during a lightning strike.

Graphite/epoxy composite structures are much less conductive than the conventional aluminum aircraft structures, both electrically and thermally. Laboratory lightning test results have indicated that a lightning strike on an improperly designed graphite composite structure can seriously degrade its structural integrity (Reference 10). New design approaches to lightning protection are required for graphite composite structures with special emphasis on low-cost, lightweight, and ease of maintenance aspects of the protective hardware designs.

The DC-9 wing box structure is primarily in a Zone 3 lightning current transfer region, as shown in Figure 3-8, except for the wing root area which is also in a Zone 2 swept-stroke/restrike region. The existing aluminum leading edge and trailing edge structures would conduct most of the lightning current flow to and from the wing tip area. The aluminum trailing edge structure would also conduct the lightning currents from the main landing gear when struck by lightning during a landing approach. A small amount of lightning current could flow in the skin panels of the composite wing box structure since they are parallel conductive paths. For a continuous box structure with no discontinuities, this limited amount of lightning current flow would be harmless to the integrity of the graphite composite panel.

The concentrated current flow along the wing leading edge and trailing edge aluminum structures could increase the energy level of the lightning-induced transients in the wiring circuits along the front and rear spars. The low shield of effectiveness of the graphite composite skin and spar structure would also increase the transient coupling energy level. This increase in the transient energy level should be determined, and unless it is demonstrated to be acceptable for critical electrical or fuel instrumentation wiring systems, additional electromagnetic shielding or transient suppression devices will be required.

The wing root area is also vulnerable to swept-stroke/restrike lightning attachments. Adequate lightning protection designs must be incorporated in the graphite composite skin panels, doors, and structural joints in this area to protect the structural integrity as well as to prevent the arc-initiated fuel ignition hazard.

The existing aluminum wing tip and landing gear installation for the composite wing concept is designed with adequate protection against the direct lightning strike, and the graphite composite wing box structure does not require direct protection from lightning strike in the design.

McDonnell Douglas test data indicate that graphite composite panels have electrical conductive characteristics sufficient to conduct 100-kiloampere restrike attachments and disperse large amounts of lightning currents without causing a fuel ignition hazard or significant degradation in strength (References 11 and 12). Therefore, the proposed graphite composite skin panels of the box structure would not require additional lightning protection hardware design for the swept-stroke/restrike attachment or lightning current transfer protection purposes.

Lightning test data (Reference 13) indicate that lightning attachment to fastener heads and lightning current flow through adhesively bonded or bolted joints in the fuel tank area could cause an arc-initiated fuel ignition hazard. The AFFDL-sponsored composite structure lightning protection programs (References 14 through 16) are not intended to investigate this problem. It should be noted that certain fighter aircraft incorporate fire-inerting systems in the wet wing box area and thus arcing inside the fuel tank may not be a fuel ignition hazard. Also, certain types of fighter aircraft incorporate fire-retardant systems which control the fuel ignition hazard caused by projectile penetration. However, current commercial aircraft do not incorporate these fire-inerting or retardant systems and lightning-initiated arcing inside fuel tank must be prevented.

Both the Government and industry have been conducting tests and analyses to determine the electromagnetic shileding properties of graphite composite structures (References 14 through 16). Several metallized surface protection systems such as aluminum mesh, spray, or foil systems have been developed which will provide a certain degree of electromagnetic shielding. When evaluating these protection systems, special consideration must be given to the weight/cost penalties and associated manufacturing and maintenance problems. These considerations become very significant for large-area applications such as wing box structure in commercial aircraft.

The electromagnetic shielding required for certain critical electrical wiring components may also be provided by local shielding or by using transient suppression or filter devices. These protection devices are usually not required in an aluminum wing box. The requirement for using these protection devices as well as their design criteria should be determined for each critical wiring circuit through analysis or test.

A lightning test program should be conducted to investigate the fastener installations in the wing root fuel tank area vulnerable to swept stroke/restrike lightning attachments. Critical design data are needed for preventing high-energy-level arcing from occurring inside a critical fuel vapor area when lightning strikes the exposed fastener heads.

An analysis and test program should be conducted to investigate a possible requirement for local shielding or the use of transient suppression devices for protection of critical wiring or components located in the graphite composite wing. A tradeoff study should also be conducted to evaluate the current metallized surface shielding protection systems versus the use of local shielding and/or transient suppression devices. The study should consider shielding effectiveness, weight and cost penalties, and ease of manufacture and maintenance.

Static Electricity - Aircraft can accumulate static electric charges by triboelectric charging when operating in an environment of precipitation (Reference 17). This charge raises the aircraft potential until it reaches a critical value at which corona discharges take place at high gradient points of aircraft. These corona discharges consist of a series of short current pulses that, when coupled into aircraft communications and navigation systems, can cause radio interference known as p-static (Reference 18). Other sources of p-static include arcing between isolated skin panels and streamers or St. Elmo's fire over dielectric surfaces. The p-static problem can seriously affect the performance of antenna systems, such as Loran, ADF, HF, and VHF. The p-static problem is usually controllable through the installation of a p-static discharger system on aircraft extremities, adequate electrical bonding practices in skin panels, and sometimes antistatic coatings over dielectric surfaces (Reference 18).

Static charges can also be accumulated inside the fuel tank through fuel sloshing during flight or during refueling. Arcing can occur inside the fuel tank between fuel probe and fuel surfaces and the fuel tank wall due to this static charge buildup. However, the energy level of these arcing activities inside the conventional aluminum fuel tank is usually well below the threshold to cause a fuel ignition hazard.

Aircraft skin panels are subject to static charge buildup caused by particle impingement of snow, ice, rain, sand, and the like. For conventional aluminum aircraft, the static charge deposited on the aluminum skin panels quickly drains through the surface coating to the metal skin and is distributed throughout the metal parts of aircraft. There has been concern that graphite composite skin panels may slow down this charge redistribution process due to its lower conductive properties, thus causing p-static problems. However, Douglas has demonstrated that graphite composite skin panels have adequate conductivity to dissipate static charges without causing p-static interference (Reference 10).

Another critical consideration associated with the application of graphite composite structures is the static charge-initiated arcing inside a composite fuel tank. It is known that static charges accumulated inside fuel tanks during fuel sloshing or refueling have initiated arcing inside the tanks across fuel surfaces to fuel probes and tank walls. However, the energy level of this arcing activity is below the fuel vapor ignition level and this arcing has not posed a problem in the conventional aluminum aircraft. For a graphite/epoxy composite fuel tank construction, the energy level of the arcing activity might be greatly increased due to the lower conductivity of graphite composite structures, the nonconductive adhesive bonding process frequency used, and the special sealing coatings. Graphite composite structures, although they possess good electrical conductive properties, are nevertheless 1000 times less conductive than aluminum structures. This higher-energy level of arcing activity should be investigated as a possible fuel ignition hazard.

General Dynamics has conducted a preliminary investigation of the fuel electrification problem associated with a graphite composite fuel tank (Reference 13). The study concluded that a graphite composite fuel tank will behave in a manner similar to an aluminum fuel tank in dissipating static charge buildups. The detailed description of the General Dynamics test program is not available at this time. It is reasonable to believe that a composite wing fuel tank design for commercial transport is different from the composite fuselage fuel tank design simulated by General Dynamics.

MATERIALS AND PRODUCIBILITY ISSUES

Ten issues have been identified in the category of Materials and Producibility Technology (see Figure 3-2). Of these, only the nondestructive inspection methods represent a key issue critical to the success of the program. An assessment of the 10 issues follows.

Material Selection

A broad base of structural advanced composite materials now exists. The great majority of these materials are supplied as "prepregs" – resin in a partially polymerized state (B-stage) preimpregnated in structural fibers to a controlled amount and a desired degree of advancement. Some are available as "wet layup" where the resin is applied to the fiber as it is applied to the part, as in filament-winding or pultrusion. Epoxy resins are the primary matrix materials, with some polyimides and phenolics available. Graphite fibers predominate, although boron and Kevlar find some applications.

The current suppliers of graphite/epoxy prepregs and a brief summary of some key properties are shown in Table 3-3. There are multiple sources of both resin and fiber although properties will vary somewhat between some combinations. Table 3-4 indicates the major resin manufacturers and Table 3-5 the graphite fiber manufacturers.

Efforts are being made to improve the existing fiber/resin systems. Each of the prepreg firms has in-house programs oriented toward improvement of one or more characteristics of his current products. In addition, there are several DOD and NASA-funded programs to directly or indirectly improve product performance, processing, or cost. Brief summaries of some of these programs are given in References 19 through 29.

The final material selection will be made at the start of the wing program. However, it is intended to select the material from state-of-the-art systems available at that time.

Aggressive Aircraft Environmental Resistance - The materials selected must resist the aircraft environment, both in flight and on the ground, for

TABLE 3-3

COMPARATIVE GRAPHITE/EPOXY PREPREG PROPERTIES

MATERIAL	MANUFACTURER	TENSILE STRENGTH MPASCAL (PSI x 10^{-3})	TENSILE MODULUS GPASCAL (PSI x 10^{-6})	COMPRESSIVE STRENGTH MPASCAL (PSI x 10^{-3})	COMPRESSIVE MODULUS GPASCAL (PSI x 10^{-6})	SHEAR STRENGTH MPASCAL (PSI x 10^{-3})
5208/T300	NARMCO	1516.9 (220)	137.9 (20.0)	1516.9 (220)	131.0 (19.0)	122.7 (17.8)
5208/T6300	NARMCO	1447.9 (210)	137.9 (20.0)	1447.9 (210)	131.0 (19.0)	117.2 (17.0)
5208/CELION3000	NARMCO	1647 9 (239)	151.0 (21.9)	1458.6 (213)	140.6 (20.4)	124.9 (18.1)
5208/CELION 6000	NARMCO	1654.7 (240)	147.5 (21.4)	1516.9 (220)	151.7 (22.0)	124.1 (18.0)
976/T300	FIBERITE	1606.5 (233)	148.2 (21.5)	–	–	117.9 (17.1)
E788/T6300	U.S. POLYMERIC	1790.5 (261)	159.9 (23.2)	1689.1 (245)	124.1 (18.0)	109.5 (15.9)
F263/T300	HEXCEL	1413.4 (205)	141.3 (20.5)	1413.4 (205)	136.5 (19.8)	120.7 (17.5)
CE9009/T300	FERRO	1489.3 (216)	139.9 (20.3)	1516.9 (220)	134.4 (19.5)	104.1 (15.1)
3501-6/AS	HERCULES	1585.8 (230)	137.9 (20.0)	–	128.2 (18.6)	120.6 (17.5)

TABLE 3-4
MAJOR RESIN MANUFACTURERS

EPOXY	POLYMIDE	PHENOLIC
DOW CHEMICAL CIBA-GEIGY SHELL	DUPONT CIBA-GEIGY GULF OIL RHODIA MONSANTO	REICHHOLD MONSANTO CIBA-GEIGY

TABLE 3-5

GRAPHITE FIBER MANUFACTURERS

TYPE	PRODUCT	MANUFACTURER
HIGH MODULUS, HIGH STRENGTH	THORNEL T300	UNION CARBIDE
	AS, HTS	HERCULES
	CELION	CELANESE
	TYPE II	MORGAN
	3T, 4T	GREAT LAKES CARBON
	HI-TEX	HITCO
	PANEX	STACKPOLE
VERY HIGH MODULUS, MODERATE STRENGTH	GY 70	CELANESE
	T75	UNION CARBIDE
	TYPE I	MORGAN
	HMS	HERCULES
	5T, 6T	GREAT LAKES CARBON

the planned lifetime without degradation of material strength properties. Assurance of this capability will require testing during and after exposure to a variety of aggressive environments. Much of this testing has already been started. Some engineering judgment will have to be exercised since real-time testing for multiple lifetimes would not permit the introduction of new materials in a timely manner. Extrapolation of ongoing tests based on the test results and of the flight history of current composite hardware would appear to be necessary for those cases where an accelerated test is not available. Some of the effort now going on is discussed in References 19, 20, 22, 23, 26 and 28 through 35.

It will be instructive to list and discuss some of the more critical environments.

1. Fluids

The commercial transport wing will come in contact with the following fluids: jet fuel, salt spray, hydraulic fluid, water, oil, and de-icing fluid. The effect of jet fuel must be assessed since the wing is wet. No problem is expected with the fuel itself. However, there is some evidence of microbial attack on the structure from impurities common to the fuel after extended flight. This condition and a suggested program are covered in another section of this report.

Industry standard tests for salt spray, hydraulic fluid, water, and oil do not indicate a problem with graphite composites (Reference 36). The design data program for the wing should adequately assess these environments.

2. Other Environments

A. Temperature and Humidity - There is strong evidence that the effects of temperature and humidity cannot be treated independently. The combined effects are synergistic in a direction unfavorable to performance. When graphite epoxy laminates with high moisture content are exposed to high temperatures or rapid temperature rise, the strengths decrease markedly. Fortunately, there is no evidence of irreversible damage within the envelope of commercial aircraft operating conditions. The reduction in strength at temperatures up to 93°C (200°F) appears to be minimal (less than 10 percent) even for laminates saturated with moisture when tested statically. Data are sparse on low- to high-temperature cycling combined with humidity exposure, or with superimposed load profiles. It is believed that additional investigative efforts will be necessary. Even though a real problem probably exists only for supersonic aircraft, the publicity already given to this exposure makes it a sensitive issue that must be addressed. Currently funded programs will provide much greater insight into this potential problem and into a definition of accelerated test methods (References 36 and 37).

B. Microbial Attack - The effects of microbial attack are discussed in those sections of this report dealing with the evaluation of coatings and jet fuel.

3. Interaction

It would be highly desirable to develop tests that would provide data on logical combinations of several of the critical environments. Superposition of load profiles would also be helpful.

Impact Resistance - Recent studies at NASA and elsewhere have shown that impact damage, undetectable visually, can cause drastic reductions in compression strength, as much as 40 percent. Consequently, corrective measures must be taken. There are some currently funded programs addressing this problem, but more are needed. The solution may come through one or more of the following:

1. Use a more ductile resin.
2. Improve the resin-fiber interface bond.
3. Use combinations of different fibers (e.g., glass or Kevlar).
4. Alter the pattern (e.g., more transverse fibers).
5. Alter the design to provide crack stoppers.
6. Sacrificial surface layer (rubber, fiberglass, Kevlar, screen, etc.).
7. Combinations of the above.

In addition to the above, close attention must be given to minimum-gauge requirements for typical impacting objects such as gravel or dropped tools. Since this is primarily a design consideration, it is covered in that section of the report. Significant effort is now being expended on these requirements, as indicated in the literature. See, for example, References 21, 33, and 38 through 44.

Smoke and Flame Resistance - There are no special requirements for smoke or flame resistance in wing structure at present. There does not appear to be any significant activity in this direction. Should such requirements appear, the ongoing efforts for interior structure would provide insight into the material modifications necessary. However, current hot-melt graphite/epoxy composites are self-extinguishing.

Material Variability - Currently available graphite/epoxy prepregs exhibit considerable variability. Following are some of the problem areas:

1. Variation in resin content.
2. Variation in degree of advancement.
3. Poor fiber alignment (particularly in unidirectionally woven fabrics).
4. Nonuniform wet-out of the resin.
5. Weaving defects, splices, etc.
6. Variation in strand count.
7. Possible variation in resin composition.

Variability in the testing makes evaluation of many of the above particularly difficult. Work is in progress in some tests (References 45 through 47).

While these variations are not critical to design or fabricability, they will affect design efficiency and manufacturing cost.

A composite wing technology design data test program that will be observed and approved by the FAA should be sufficient to verify material conformance in all of the foregoing areas except microbial attack and impact resistance. Development programs are recommended in these two areas.

Electrical Hazard Evaluation - In addition to the preceding requirements, the material used in commercial aircraft may have to meet new standards for reducing the risk of dispersion of bare fibers in the event of fire and impact. This problem is currently being assessed by a risk analysis to ascertain the expected loss and by considering what material changes would be needed to alleviate or eliminate the problem, if there is one. There are several contracts in progress or being bid for work in this area. It is apparent that a material change would necessitate a large-scale test program to cover not only the environmental characteristics but the structural characteristics as well. However, this potential problem relates to all graphite/epoxy on the aircraft and is not unique to the wing.

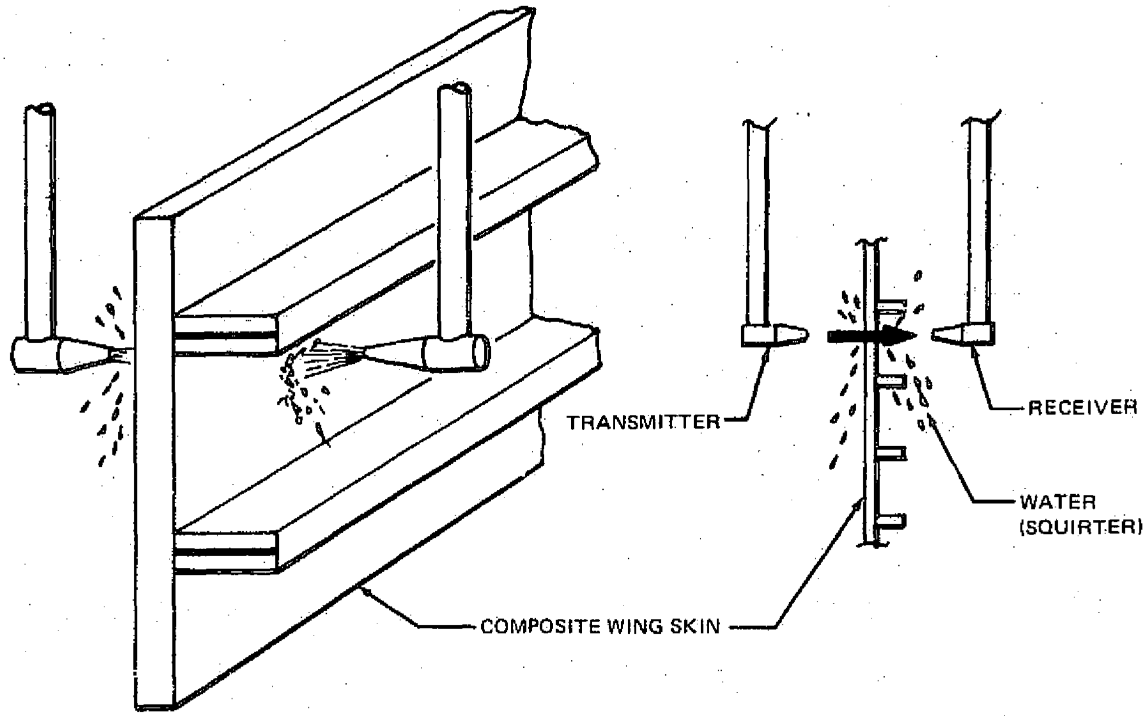
Nondestructive Tests

Nondestructive testing or inspection of graphite/epoxy composite structure presents many challenges when a large structure such as the composite wing is considered. The wing structure must be inspected after fabrication for defects such as voids, cracks, porosity, and delaminations. Experience in inspecting graphite composites has shown that such defects can be detected using x-ray radiography, ultrasound, and dye-penetrant (References 48 and 49).

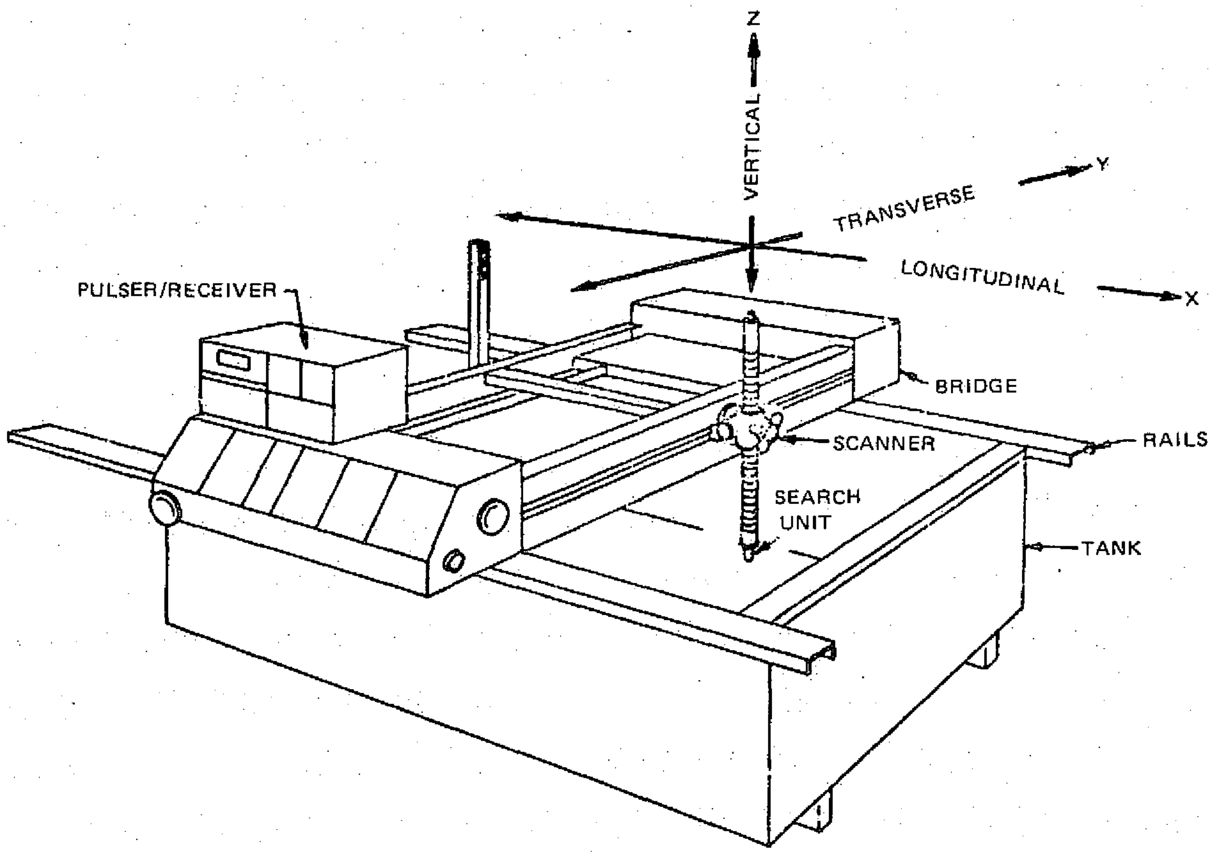
The selection of nondestructive test method is generally based on part geometry and composition, potential minimum defect size, location, orientation, and availability of test equipment. Very often, more than one nondestructive test method is used because different conditions or defects are revealed by each method.

The DC-9 composite baseline concept is to be made from graphite/epoxy composite laminates. This design concept indicates that ultrasonic C-scan is the most promising method for inspection of the wing skin and spar cap laminates. This concept is described in Figure 3-9.

Special and costly automated C-scan equipment will need to be purchased to inspect the 15.24-meter (50-foot) long skins and spars. Typical defects that can be detected by ultrasonic C-scan inspection in graphite/epoxy laminates are shown in Figure 3-10. X-rays will also be used in identifying foreign objects detected by ultrasonics. It is envisioned that X-ray work will be done using portable equipment and shields.



(A) THROUGH-TRANSMISSION TECHNIQUE WITH SQUIRTERS



(B) PULSE-ECHO IMMERSION TECHNIQUE

FIGURE 3-9. TYPICAL ULTRASONIC C-SCAN INSPECTION TECHNIQUES

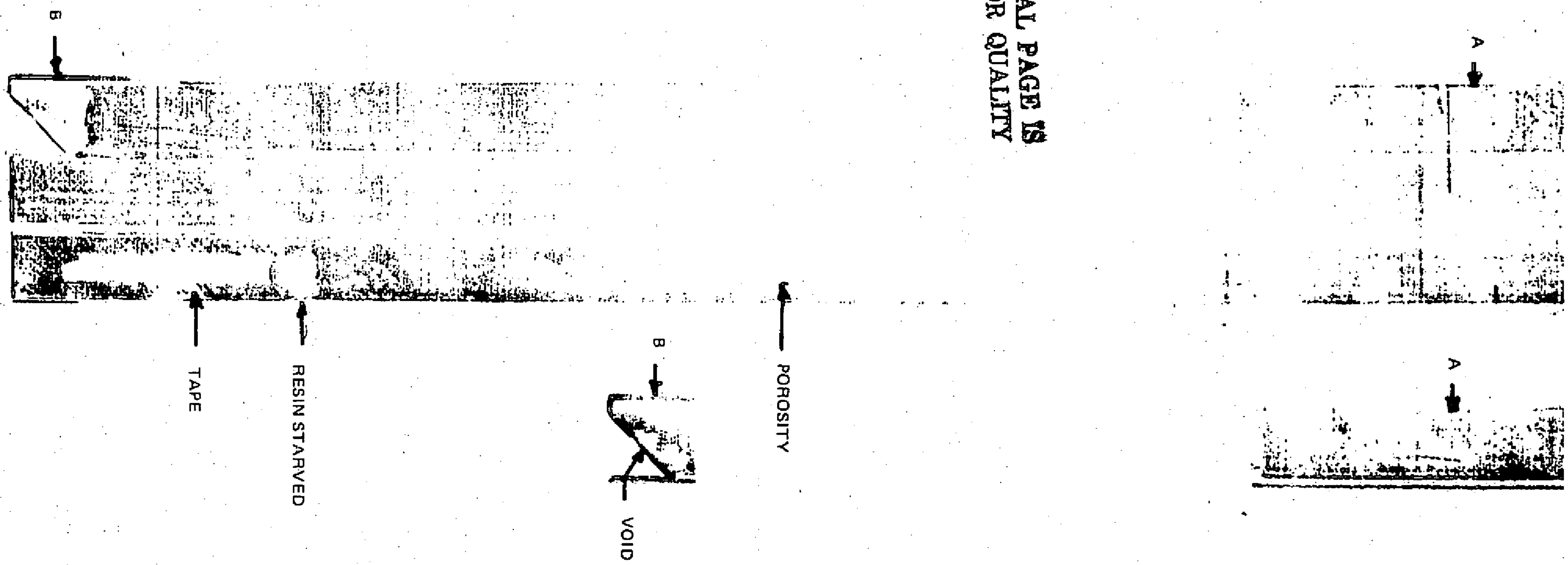


FIGURE 3-10. ULTRASONIC C-SCAN RECORDINGS OF GRAPHITE/EPOXY COMPOSITE STRUCTURE

Bond testers such as the Fokker and NDT-210 will be required to determine the quality of adhesive bonds. Adhesive bonds are somewhat difficult to inspect if the parts are cocured and bonded, and if delamination or porosity is contained in the graphite composite. A defect in the composite laminate will prevent the ultrasonic energy from reaching the bond joint; hence, the quality of the bond cannot be determined. Also, graphite-to-metal bond joints are more difficult to inspect than graphite-to-graphite bond joints because the acoustic impedance mismatch is greater for the graphite-to-metal joint, which reduces the signal amplitude ratio between a bond/unbond condition.

In any case, special built-in deflect standards will need to be fabricated to calibrate the bond testers prior to inspection for the cocured skins and titanium doublers. Nondestructive test technology for bond joints should be developed during other ACEE programs and be available for the wing program.

For process control during fabrication, the dielectric test method shows promise in evaluating or monitoring the curing of the resin (Reference 50). The degree of cure is dependent on a time versus temperature relationship, B-stage condition of the resin and on thickness and heat sink from tooling adjacent to the composite material. A critical variable of the cure cycle is the correct time to apply pressure. The optimum time for applying pressure is when the resin reaches a certain viscosity so that it will flow under pressure, all volatiles having escaped, and consequently produce a void-free structure. This process is presently being studied under production conditions and should be ready for use on the wing program.

One of the most common flaws encountered in graphite/epoxy composite laminates is interlaminar porosity. Both theoretical predictions and experimental evidence show that there are decreases in mechanical properties in the presence of voids (Reference 51). Data show that the interlaminar shear strength of composites decreases by about 7 percent for each 1 percent of voids, up to at least 4 percent void content. Other resin-critical mechanical properties may be affected to a similar extent. Hence, essentially void-free composites are necessary for primary aircraft structures.

All the methods used for quantitative measurement of void content have limitations. It is doubtful, at the present time, whether an accuracy much better than void content ±0.5 percent can be attained with any of the available techniques (References 52 and 53). There is thus a need for an accurate method of measuring void content, especially at the 0 to 1 percent level. An example of ultrasonic C-scan comparing laminate void content is shown in Figure 3-11. The laminates can be made with variable resin-starved or resin-rich areas. Void content is best measured by ultrasonic attenuation whereas neutron gauging appears as the best method for measuring resin content (Reference 54). The developments during other current ACEE programs with similar structural requirements and low-void-content laminate requirements suggest that these measurement techniques can be expected to be available by the time of the wing program.

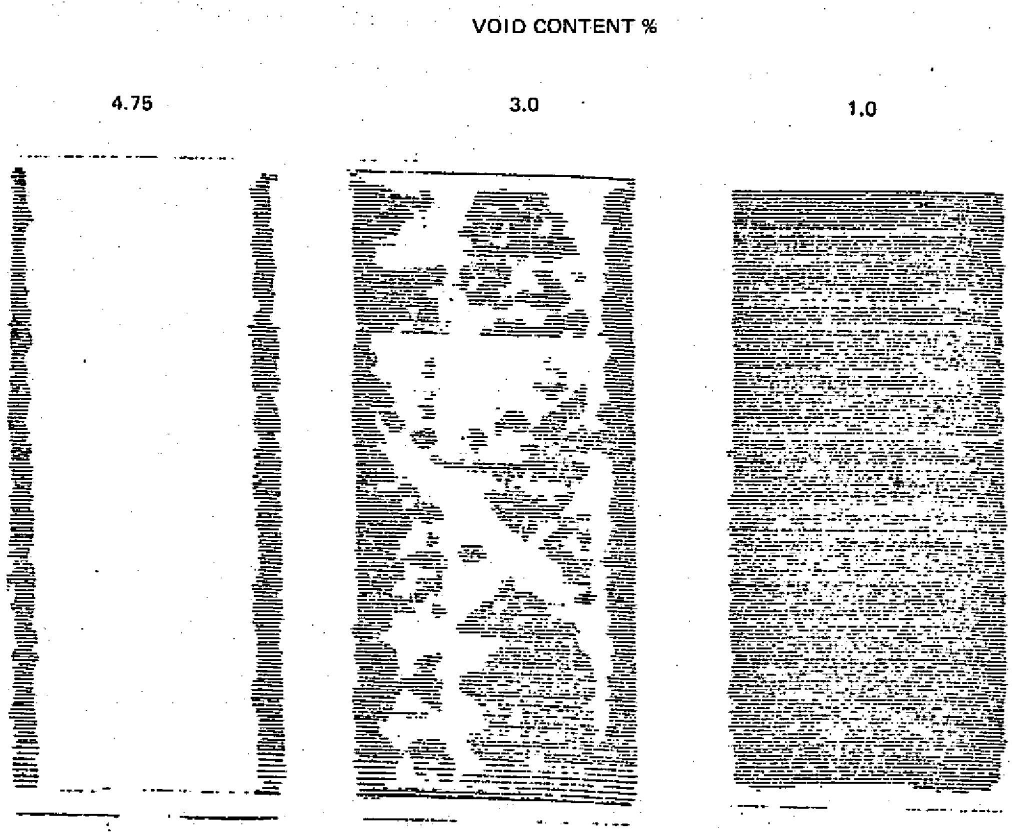


FIGURE 3-11. ULTRASONIC C-SCAN AT 2.25 MHz OF 32 PLY CARBON FIBER REINFORCED PLASTIC LAMINATE (VARIABLE VOID CONTENT) REFERENCE STANDARDS

More effort needs to be directed at investigating the effects of defects. Production parts must be inspected to some kind of accept/reject criterion and this criterion must be realistic. A major decision must be made when very large and expensive parts contain defects. If the Material Review Board elects to accept the rejected part, then it can become a problem for the personnel performing in-service inspections. During the in-service inspection, the production defects may be detected and considered as flaws developed during service. Therefore, pictorial records of useful nondestructive tests are necessary. The relationship between flaws and their effect on mechanical properties must be firmly established for primary structures. Tentative acceptance criteria have been and will be set for graphite/epoxy composite structure (Reference 54). Specimens need to be fabricated with various built-in defects measured by nondestructive test methods. The specimens will then be fatigue-tested to determine flaw growth characteristics.

Some programs have been conducted, but a relationship needs to be established for specific materials used for the wing. All these things will lead to realistic accept/reject criteria for production and in-service inspections.

To establish durability, in-service inspection of selected critical areas of the wing structure will require in-service nondestructive inspections. Parts with flaws may find their way into service and this can cause difficulty to the personnel performing the nondestructive inspections unless the inspection is being conducted in an area away from the initial production flaws or discontinuities. Copies of the rejection records would be required by in-service inspection personnel. Photographs of fabrication inspection C-scans will provide a printed record for reference during in-field inspection.

Early in the detail design phase, the areas to be inspected during service and the inspection frequency need to be defined. Also, the accept/reject criteria must be determined. The inspection methods must be documented and must be very specific.

Guidelines for in-service nondestructive inspections should be forthcoming from a NASA-Langley Research Center program entitled "Evaluation and Development of In-Service Inspection Methods for Grahpite/Epoxy Composite Structures on Commercial Aircraft."

There are additional problem areas connected with in-service durability of graphite/epoxy composite structure. One is aging and environmental damage to composite materials. A program has been started with Boeing to determine the environmental effects on graphite/epoxy composite structure. However, no nondestructive test method will be evaluated in this program - only mechanical tests are to be periodically performed. If graphite/epoxy composites are to be used on primary structure for commercial aircraft, nondestructive test methods may be required to determine the degradation of the structure as related to strength and fatigue life.

Another anticipated problem is fire damage. Aircraft structure, especially wing structure, is subject to fire damage in the area of the wheel well. The significance of fire damage must be determined. Nondestructive test methods must be evaluated to determine if they can relate the fire damage to loss in physical properties or fatigue life. Similar relationships have been established between eddy current conductivity, hardness, and loss in yield or ultimate strength in metals. This program is extensive and will require study to define the range of parameters to be examined. There are no known programs involving assessment of fire damage.

Quality Control

The structural integrity of the aircraft must be assured during its manufacture and throughout a long period of commercial service. Assurance that the necessary quality exists at all times is essential. Since mechanical testing of the final item is not a viable approach, this quality assurance must come from nondestructive means coupled with inspections and tests during manufacture and in-service use.

Quality assurance is a technological issue for the wing because of the work needed to develop efficient methods of process verification and nondestructive test standards. In addition, the effect of defects must be known so that technically sound decisions can be made as to their disposition (e.g., accept as is, repair, or reject).

The present state of the art of quality assurance for advanced composites is somewhat uneven in coverage. Some activities are well defined while others are not. The status of the various activities is given below.

1. Well-Defined Activities
   - A. Raw material control and testing
   - B. Dimensional checks
   - C. Traceability
   - D. Tool inspection
   - E. Equipment certification
   - F. Procedural control.
2. Activities Needing Modest Improvement
   - A. Process control
   - B. In-process quality tests other than dimensional or nondestructive
   - C. Defect standards (e.g., type, size, or location)
   - D. Layup verification
   - E. Evaluation of repairs.
3. Activities Needing Development
   - A. In-service tests
   - B. Nondestructive tests
   - C. Nondestructive test standards.

All of the well-defined activities will be readily incorporated into the quality assurance plan and consequently need not be discussed in this report. Currently, programs funded at Douglas and at other aerospace firms will provide the necessary improvements for the intermediate category of activities. (See References 28, 30, 32, 33, 45, 46, 47, and 55 through 59). In addition to the efforts referenced, Douglas is currently monitoring the curing process dielectrically, evaluating the efficiency and inspectability of repair techniques, and is closely following industry efforts. The improvements needed in layup verification, defect standards, and in-process quality tests are in reducing the cost of present techniques (e.g., one-on-one inspection of layup versus black-on-white or computerized scanning). The development work on the effect of defects will help to decrease the total number of standards necessary and to minimize their complexity.

Machining of Composites

Manufacturing development in composite airframe assembly applications in the past several years has generated improved drilling, reaming, countersinking, cutting, machining, sawing, and routing methods that can produce high-quality cut surfaces. Significant machining data have been gathered from NASA and military contracts at Grumman Aerospace Corporation, Rockwell International, and Rohr Industries (References 60 through 62).

McDonnell Douglas and others have developed acceptable and useful machining technology on independent research and development and many contracted programs.

The most widely used machining and cutting technique is with a high-speed diamond wheel or abrasive cutoff wheel. This technique can be expected to be utilized in the long, straight cuts expected in the large-wing structure.

Large machines capable of supporting the great size of the wing structure and locating with precision the cutoff wheels, dust collectors, and coolants must be designed and built. Irregular cutouts are successfully made using diamond or abrasive wheel routers or diamond blade band saws.

High-quality holes are drilled using specially shaped carbide or diamond core drills, reamers, and countersink cutters with controlled feed and speed. Back side support is frequently used to minimize or eliminate "breakout." This support is usually either a hard surface such as aluminum or hard board or a ply of fiberglass that stays with the part (as part of the electrical insulation) or may be removed as a peel ply after the holes are drilled. These procedures seem appropriate for the large, contoured wing where conventional drilling jigs or fixtures are supplied.

Possible low-cost drilling or cutting techniques may be forthcoming from some of the developments in progress, such as cavitation, ultrasound, laser, water jet, or new, more conventional tool geometry concepts.

Potential Problems - There are several potential problem areas that may influence the wing problem.

1. The great size of the composite structure presents pervasive processing problems, along with the massive tools required to scale up from known machining methods.

2. Drilling and cutting methods for mixed graphite composite and metal (titanium, aluminum, or steel for local reinforcement) have not been optimized. Trouble could arise from overheating or metal chips damaging the composite areas of a hole or edge during the machining operation. The many existing contracts and the need for integral metal local reinforcement in these pieces of hardware can be expected to produce adequate machining methods.

3. The full effect of flaws in holes and edge cuts has not yet been established for the long-range durability of the structure. These investigations are now underway. The allowable hole tolerances, both for dimension and for flaws, will directly affect the costs of the drilling operations.

4. The process of machining, cutting, and drilling is a significant portion of the overall cost of a composite structure. Where a part is of marginal cost-effectiveness, the expected lower-cost machining techniques may be necessary for a design to be released for production.

5. Personnel health, safety, disposal of composite dust, and parts handling have not been a problem on current programs. For large-scale production, training will be required for personnel prior to their exposure to the shop. Suitable and economical equipment for meeting possible OSHA requirements on collection and disposal of composite dust will require careful planning.

Microbiological Fuel Contamination

Microbiological contamination of turbine fuels (kerosene type) in integral fuel tanks and fuel distribution systems has been a quality control problem since the late 1950s. Left unchecked, microorganisms can affect fuel quality and aircraft reliability. They suspend water and particulate matter in fuel and promote the formation of sediment and gums. Under the right set of conditions, microorganisms can eventually cause wing tank corrosion, fuel pump and filter clogging, and capacitance gage malfunctions, and can contribute to engine failure.

The problem was most acute during the late 1950s and early 1960s, particularly in humid regions of the world such as the Far East (Reference 63). Also, aircraft in the United States were found with large amounts of sludge and fungal and bacterial growth in the wing tanks (References 64 and 65).

The microorganisms that contaminate kerosene fuels have been isolated and identified by many investigations (References 63 and 66 through 69). The predominate organisms are the fungus Cladosporium resinae, the bacteria Pseudomonas, and the yeast Candida. These microorganisms utilize hydrocarbon fuels as a carbon source for energy and concentrate at the interface between the two phases of water and hydrocarbons. Water serves as an electrolyte in a corrosion cell and is an essential nutrient needed by microorganisms. It is through the medium of water that some microorganisms can change the environment to one that is corrosive to aluminum. It is thought that microorganisms can instigate the corrosion of aluminum in fuel tanks by more than one mechanism. Possible mechanisms include creation of a galvanic corrosion cell mediated by the microbial enzyme hydrogenase, establishment of a differential oxygen cell, direct utilization of the metal, and a change of the environment to one that is more corrosive by excretion of organic acids as metabolic byproducts (References 70 through 74).

Since microorganisms require water to grow, good housekeeping in fuel handling is important from the fuel distribution system to the aircraft. By allowing sufficient storage tank settling time, maintaining filter/separator equipment and regular sumping of free water from storage tanks, and using filter sumps and aircraft fuel tank drains, the fuel may be free from water and particulate matter. Often, this is not enough. Microbiological fuel tank contamination problems vary considerably from one airline operation to another. In September 1976, the Aircraft Fuel Tank Corrosion Group of the Coordinating Research Council (Reference 75) prepared and distributed approximately 140 questionnaires to 110 airlines, of which 86 were foreign. The questionnaire was aimed at determining the economic impact resulting from microbial contamination of fuel in jet aircraft and the airlines' experience with fuel biocides. Of the 41 airlines responding, most have experienced microbiological fuel contamination problems and only 18 of the responders do not use fuel biocides.

In addition to good housekeeping practices, which are sometimes beyond the airline's control, there are many other variables. Some of these factors are the type of aircraft in a fleet; how well the aircraft tanks drain and if they have a water scavenging system; the geographic routes and flight schedules; fuel tank inspection practices and frequency; and use of fuel biocides.

Thus, if aircraft fuel tanks are to be constructed of graphite/epoxy composites, the possible effects of periodic microbiological fuel tank contamination in localized areas must be considered. Microorganisms usually attack plastic polymers with their extracellular enzymes and utilize the carbon molecules in the plasticizer for their metabolic processes. Usually, this type of attack results in visible pitting with subsequent loss of mechanical strength and loss of flexibility (Reference 76). However, since a composite such as graphite/epoxy contains a thermosetting resin system brought to a cured state by heat activation and contains no plasticizers, direct utilization of the plastic is not likely.

When microorganisms grow in jet fuel, they oxidize the aliphatic fractions in the kerosene fuel most rapidly, and form water-soluble fatty acids, higher alcohols, aldehydes, and other intermediates. Some of the lower fatty acids (organic acids) such as acetic, formic, propionic, or butyric may in time have a deleterious effect on the epoxy resin since they are less resistant to these types of materials. A preliminary study was conducted on the resistance of a graphite/epoxy composite to microbial attack in a fuel/water environment (Reference 77). It was shown to be relatively unaffected by short-term (14 days) microbial attack but, in a two-month exposure test, there was an indication of microbial degradation. Thus, the long-term effects of microbiological fuel tank contamination remain uncertain.

Two military aircraft programs, the F-18 and the AV-8B, feature integral wing fuel tanks with graphite/epoxy structure. It is expected that much of the microbiological fuel contamination issue will be resolved by these programs. Some additional testing is recommended with regard to materials and design concepts typical for commercial transport aircraft.

Fuel Tank Sealing

The seams, joints, and mechanical attachments in an integral fuel tank box structure require a precision sealing operation to preclude fuel leakage. This is true for the present aluminum construction and will be true for the proposed composite wing construction.

Standard procedures developed for sealing the seams and joints that form the boundary of integral tanks and fastener sealing have been highly successful on the DC-9 and DC-10 aluminum wing airplanes. A combination of a polyurethane base internal coating for corrosion and microbic control and a polysulfide base sealant has been in use for well over 10 years and no problems have been encountered in production or in service. The polysulfide base sealant has elongation in excess of 200 percent, which allows it to accept the relative movement of the structure under load.

The same polyurethane interior coating and polysulfide sealant materials and processing system are in current use with the graphite/epoxy integral wing fuel tanks on the F-18 and Harrier aircraft at McDonnell Douglas Corporation. Careful, proven techniques must be used to achieve reliable wing tank sealing. In general, the following procedures are recommended:

1. Coat the parts of the structure which form the fuel tank boundary and the parts in the boundary with a polyurethane-based coating (MIL-C-27725) prior to assembly or installation.

2. Prepare the polyurethane surface for bonding and assemble the joints or seams that form the fuel tank boundary with elastomeric polysulfide-based sealant (MIL-S-81733) on the contacting surfaces (faying surfaces). See Figure 3-12.

3. Apply a fillet of elastomeric polysulfide-base sealant (MIL-S-8802) after assembly to all joints and seams that form the tank boundary. See Figure 3-12.

4. Install all fasteners through the fuel tank boundary with polysulfide-base sealant in the hole and in the countersink or under the head and both surfaces of all washers. See Figure 3-12.

5. Seal all fasteners by the same techniques through graphite/epoxy composite parts or assemblies regardless of whether they penetrate the fuel tank boundary or not (corrosion protection).

6. Seal all faying surfaces of all metallic parts that contact the graphite/epoxy parts or assemblies with polysulfide-base sealant.

7. Pressure-test the completed tank to check for 100-percent seal.

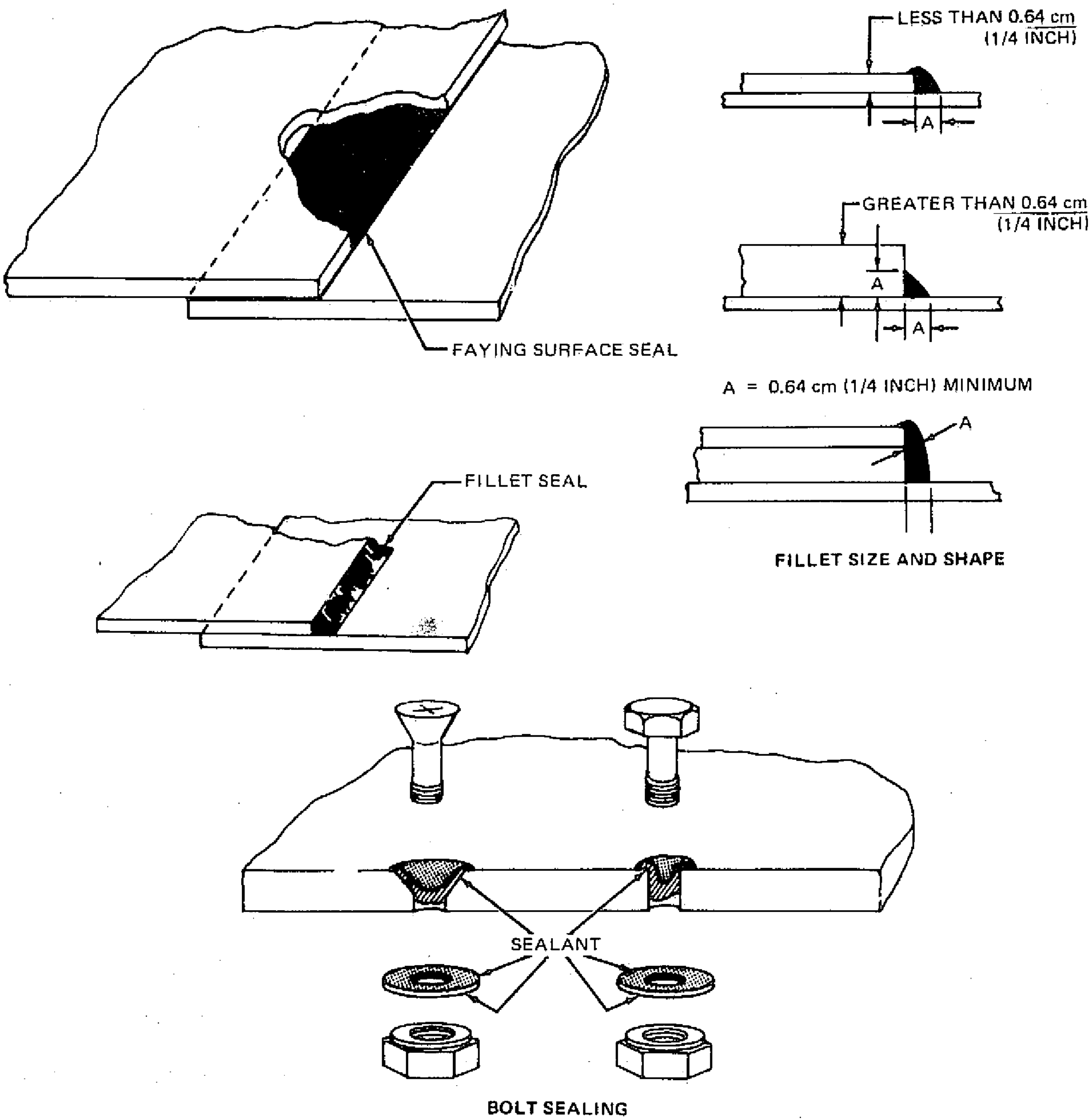


FIGURE 3-12 FUEL TANK BOUNDARY SEALING

Adhesive Bonding

Increased use of adhesive bonding has been seen with each new aircraft, both in amount of area adhesive-bonded and in the size of the bonded assembly. Examples are fuselage bonding on the L-1011 and B747 wide-bodied aircraft. This increased size of bondment has complicated the requirements for the selection of an adhesive system. An adhesive has been selected primarily for its mechanical properties such as tensile lap shear, peel, and creep. In addition, the adhesive's resistance to environmental exposure such as temperature-cycling during its service life is considered. The adhesives must also resist immersion in fluids such as engine oil, hydraulic fluid, jet fuel, water salt spray, and deicing fluid, and high humidity. Large-area bonding has required that adhesive formulations be modified to give higher flow characteristics to allow volatiles and air to escape from the bondline and provide acceptable bond joint strength. Additional types of testing have been conducted combining cyclic or sustained loading in hostile environments. This type of testing was done when it was determined after evaluating service type failures that a hot, wet environment was one of the most adverse conditions an aluminum-bonded structure could be subjected to.

Present adhesive resins most widely in use are modified epoxies designed to cure at temperature ranges of 121°C (250°F), 149°C (300°F), or 177°C (350°F). These materials are an intermediate or low-modulus system normally having high peel characteristics. One example of this type of material is FM300. This material has been selected for use on the F-18 aircraft both on metal and composite adhesive bonding. The material cures at 177°C (350°F) and is compatible with the epoxy resin systems used in present epoxy/graphite prepregs. FM300 can operate at a service temperature of 140°C (300°F), which was a requirement for the aircraft. Adhesives can be used with precured laminates as well as in cocured prepreg epoxy/graphite and titanium details (References 78 through 81). If higher operating temperatures are necessary, there are epoxies such as FM400 which also cures at 177°C (350°F) but can operate at higher service temperatures

[216°C (420°F)]. Polyimide adhesive systems such as FM34 can be used with epoxy/graphite and polyimide/graphite prepregs for temperatures in excess of 204°C (400°F).

Surface Preparation - This operation is one of the most critical in the bonding sequence. Many efforts have been made to develop optimum surface preparations. In recent years, it has been discovered that inadequate surface preparations have led to premature service failures of adhesive-bonded aluminum panels that operated in areas of high humidity. Programs such as the Air Force Primary Adhesive-Bonded Structure (PABST) (References 82 through 84) have shown the improved surface preparations such as the optimized phosphoric acid anodize initially developed by Boeing and the optimized chromic acid anodize initially developed by Forest Product Laboratory gave increased environmental resistance to an adhesive-bonded aluminum panel.

Surface preparations of cured epoxy/graphite laminates have been studied (Reference 83). These methods are mainly mechanical, such as grit blasting or hand abrading, or the use of a peel fabric which can be removed after a laminate is cured to produce a clean, prepared surface or bonding.

Surface preparations for titanium have been studied by many agencies, with many variations in the methods of preparing the surface. The most widely used method is the Pasa-Jell 107 process, employing a hydrofluoric, chromic, nitric acid solution. This is currently in use on the F-14, F-15, and F-18 fighter aircraft for titanium adhesive bonding. This system requires that material be mechanically cleaned as well as chemically treated. The environmental durability of this system and other processes has not been evaluated to the extent of aluminum bond. A Company-sponsored program is presently underway at Douglas to evaluate the environmental durability of this system and several others such as phosphoric acid anodize and a chromic anodize developed at Boeing. Additional programs are presently being proposed by the Air Force to evaluate the compatibility of various adhesive systems and titanium surface preparations.

Processing - Pressure and rate of heat-up and cure temperature are critical in obtaining the optimum properties of the adhesive. A positive pressure can be applied by several methods: in an envelope membrane which is placed in a heated pressure vessel (autoclave); by a press; and by mechanical means such as clamps, springs, or rubber bladders. Vacuum pressure is undesirable for curing many epoxy adhesives due to the expansion of the volatiles in the adhesive resulting in a porous bond joint and reduced properties. The autoclave pressure range can be from 69 to 1034 kPa (10 to 150 psi). Wide-area bondments require the highest pressures, 69 GPa (100 psi) or higher, to aid in forcing the air and volatiles from the bond joint. The heat-up rate of the adhesive, if cocured, must be compatible with the epoxy-graphite laminate cure cycle [from 0.50 to 5.5°C (1° to 10°F) per minute]. Most epoxy adhesives such as the FM300 are compatible with this, epoxy/nylon adhesives being exceptions. Epoxy/nylon adhesives require a high rate of heat-up to allow the adhesive to flow properly [approximately 4.8 to 6.5°C (7° to 12°F) per minute]. Laminates having a large mass cannot meet this type of heat-up rate unless integral heated tooling is made available.

Resins in the adhesive system must be compatible with the resin matrix in the graphite prepreg for purposes of cocured prepreg laminates to metals or precured laminates. Adhesives must be able to cure in the same range. Bonding of dissimilar materials such as aluminum or titanium to graphite/epoxy requires development of a cure cycle that can help relieve the stresses induced by the difference in the coefficient of thermal expansion of the dissimilar materials. The selection of a low-modulus adhesive helps to alleviate this problem.

Current Programs - The current ACEE programs, the vertical stabilizers at Douglas Aircraft and Lockheed, the horizontal stabilizer at Boeing, and the military programs at McDonnell Aircraft all are considering major cocured titanium to graphite/epoxy structural joints. This means that sufficient information must be obtained to assume long-term durability of this type of construction in aggressive aircraft environments. Vacant spots in current knowledge, such as PABST type temperature/humidity load conditions, can be expected to be filled in existing programs which will all be completed prior to a wing program. Although load levels may be higher for major wing structure, the stress level along the bond lines (load transfer at bond lines) is expected to be similar.

A major wing program will have its own specific design, subcomponent fabrication, and test program. The baseline technology should be available concerning adhesive strength, environmental effects, durability, thermal expansion, cocuring, and secondary bonding.

Insulation and Corrosion Control

Graphite/epoxy composites have been found to cause accelerated galvanic corrosion of the major aircraft structural metals, including the aluminum, steel, and certain stainless steels. The highly corrosion-resistant metals, such as titanium and high-nickel alloys, are not significantly attacked. The galvanic attack results from the fact that when two dissimilar metals are electrically connected in the presence of moisture, an electrical current flows from the metal with the least corrosion-resistance through the moisture path to the metal with the greatest corrosion-resistance. The current flow accelerates the rate of metal removal, or corrosion, of the least resistant metal. The metals having the greatest corrosion-resistance are known as noble, or cathodic. The active metals are known as active, or anodic metals. A classic example of accelerated galvanic corrosion is the electrolysis of steel boat hulls.

Studies have determined that graphite/epoxy composites react in the galvanic series of metals as a noble, or cathodic, metal (the metal, not the graphite/epoxy is attacked). When the galvanic potential is measured against a standard colomel or hydrogen electrode, the graphite/epoxy is found to be considerably more noble than titanium alloys and high-nickel alloys such as Inconel and Rene' 41. Only gold and platinum were found to be cathodic to graphite.

In order to assess the potential damage to metals coupled to graphite/epoxy, tests were run in which the actual corrosion currents were recorded. Since the amount of current flow is a direct measure of the amount of oxidized (corroded) metal, galvanic current corrosion tests are considered to be representative of service conditions.

Grahpite/epoxy composites have undergone extensive corrosion tests to determine the effect of coupling to aircraft structural parts and fasteners. Typical investigations are reported in References 85 through 88, by Air Force and Navy Laboratories. McDonnell Douglas has conducted static salt-fog environmental exposure tests of various metal-fastener combinations with graphite/epoxy which confirm the results of these investigations (see Reference 89). The test results showed that aluminum, steel, and lower-alloy, corrosion-resistant steels would be severely damaged if coupled directly to graphite/epoxy in a corrosive environment. However, titanium alloys and the high nickel-chrome alloys such as Inconel and Rene' 41, due to their high inherent corrosion resistance and surface passivity, are not prone to galvanic corrosion.

Advanced design techniques must be used to prevent accelerated galvanic attack of aircraft structure by graphite/epoxy. In general, the following procedures should be implemented:

a. Eliminate all crevices and traps between graphite/epoxy and metals by extensive faying surface sealing.

b. Install all attachments and inserts with wet sealant.

c. Use corrosion-resistant fasteners.

d. Apply primer and topcoat to structural parts before assembly.

Exterior Coatings

Graphite/epoxy composites require organic coatings for the following reasons:

1. Protection of the composite from ultraviolet degradation.
2. Retard moisture absorption in the composite.
3. Customer color preferences - aesthetics.
4. Protection from rain, hail, and dust erosion.

The aircraft industry has considerable experience in painting graphite/epoxy composites, both on an experimental and pilot production scale. One of the applications at Douglas has been the DC-10 composite rudder. The production rudders have been successfully coated with conventional aircraft coatings, i.e., epoxy polyamide primer and linear diisocyanate cured polyurethanes. No new development of coating systems is anticipated other than a minimal effort to determine the optimum surface preparation to ensure long-term paint adhesion.

The main surface preparation effort will be to determine if unique problems occur in removal of mold release materials used with the composites. Otherwise, surface preparation is expected to be the same as used for years on glass fiber-epoxy laminates. This entails thorough cleaning with a detergent scrub and solvent scrub to remove mold and surface contaminants and then abrading the surface with abrasive pads or sandpaper, filling any pinholes with epoxy putty materials, and surface-smoothing with epoxy smoothing compounds.

Coatings have been discussed with other aircraft manufacturers who have been building graphite/epoxy composites. Two of the other manufacturers have indicated no problems when coating their composites with epoxy primer and polyurethane coatings, as Douglas has been doing. One manufacturer indicated there were problems in obtaining adhesion. Two manufacturers indicated problems in obtaining sufficient moisture protection of the composite in their skin honeycomb application.

Rain-erosion-resistant polyurethane coating systems will be required if leading edges are constructed of graphite composite, Kevlar composite, or glass fiber laminates. The coating systems will be elastomeric urethanes similar to those specified by MIL-C-83231. The same rain-erosion-resistant coating and methods of application used on all fiberglass leading edge details and fiberglass radomes will be prescribed for applicable structural composite details.

Stripping of organic coatings from the composite wing will eventually be required after the wing has been in a normal service environment. Surface coatings may also need to be removed for repair of damaged structure while still in the manufacturing facility or while in later service. It is expected that strippers currently used to remove polyurethanes from aircraft structures may result in damage to the composites. Stripper development or development of mechanical techniques or water fan jet abrasion may be involved, but at a moderate expense, and can be expected to be resolved as an issue on current industry pilot production programs.

Low-Cost Fabrication

The manufacturing costs of graphite composite structures must be minimized if it is to be competitive with conventional metal structures. A well-coordinated design, processing, tooling, and manufacturing team, where the influence of each discipline is considered from initial concept of design, can greatly influence the cost of the end product. Most aircraft firms now follow this team design synthesis concept.

A number of low-cost development programs have either been completed or are proceeding in a continuing effort. A few of the most promising low-cost development programs that perhaps may have direct influence on large composite wing design and fabrication are reviewed.

1. Fabrication Guide

   The Air Force Materials Laboratory has, on contract to Rockwell International Corporation, the task of preparing an Advanced Composite Design Guide (Reference 90). The AFML has also contracted with Lockheed, GA, for the preparation of a Structural Fabrication Guide for Advanced Composites (Reference 42). These documents present an excellent summary for design, tooling, and manufacturing concepts. They will save original design time, be valuable for reference, and present many useful low-cost directed manufacturing methods.

2. Reduce Part Count

   There is general agreement in the industry that substitution of graphite/epoxy for metal components on a detail part-by-part concept cannot be cost-effective. The direction for composite fabrication and design is to minimize the number of individual details. The use of honeycomb construction, to reduce part count has, for example, been successfully applied at Rockwell International, Northrup Corporation and Grumman (References 91 through 93).

   Minimizing the number of manufacturing steps, such as autoclave temperature/pressure cure cycles, to produce a finished part, results in lower fabrication costs. Northrup has produced cocured and bonded honeycomb composite skin sandwich construction (Reference 92). Douglas

Aircraft Company has fabricated the DC-10 upper aft rudder which has structure of skins, spars and ribs, all solid laminate, but all integrally cured in one curing cycle (Reference 94).

3. Automated Tape Layup Machine

A number of manufacturers have explored the economics of using numerical control automatic tape layup machines for cost reduction. General Dynamics, LTV, Boeing, Lockheed, and others are evaluating these machines (References 95 through 97).

Their advantages are: (1) rapid, automated, precision layup; (2) minimal human error; and (3) automatic documentation.

Their disadvantages are: (1) high initial cost; (2) large production order of parts required (usual practice for commercial aircraft is a block-by-block release for production); (3) limits on doubler buildup; and (4) requires higher uniformity of prepreg tape (downtime to remove local defects in tape is costly).

The cost-effective applications for automatic tape layup machines should be established by these manufacturers. The machines will have their place in industry where large numbers of repeat parts are required. The detail design of the wing and subcomponent elements of the wing will determine their possible use.

4. Pultrusion

Subcomponent details similar in shape to many roll-form details can be formed by a continuous automated pultrusion process. Rolls of material (with prepreg or wet resin impregnation) can be pulled through reducing dies to a final cross section and partially cured to a hard B-stage. This shape can then be handled and installed in a complex structure and cocured and bonded in a later process cycle. Boeing and Goldsworthy Engineering have performed successful development with this process (Reference 21). Mechanical properties of cured graphite/epoxy pultrusions reported to date have not been as high as the press-cured or autoclave-cured specimens. A hard B-stage pultrusion, later cured in the autoclave, does produce typical autoclave type mechanical properties.

5. Woven Graphite

Graphite fibers are woven in to many cloth configurations that vary from 95 percent unidirectional 0 degrees to 50/50 bidirectional fabric. It may also be woven at 95 percent 45 degrees fiber direction with 5 percent 0 degrees Dacron tie yarn. It can be woven in a variety of weaving styles and thicknesses. The weaving process is automatic and this tends to prepare an economical and uniform building block material. All fabricators have reported reduced layup time using woven cloth, by as much as 75 percent compared to hand layup (References 41, 92 and 98). The wide, thick woven cloth layup time may be competitive with automatic tape machine layup. The material can be pulled over the same contour and does not split, unlike unidirectional tape, particularly when the material ages or starts to dry.

The woven form of graphite offers a mechanical means of obtaining a uniform hybrid, mixed fiber content in a panel that may act as a crack stopper, to improve impact, or simply lower cost by dilution with a lower-cost fiber in a noncritical direction (References 98 and 99).

6. Low Resin Content Prepreg

Investigations by Northrup (Reference 92) and others have been successful in reducing manufacturing costs by purchasing prepreg materials with close to the desired product resin content. The excess resin bleed is not thrown away – this was previously thought necessary to facilitate removal of trapped air during layup and, further, the procedure eliminates the need for most of the bleeder cloth. This represents real savings in labor and materials. The low resin content is particularly attractive with the woven materials where splitting is not a problem.

7. Hybrid Materials

Dupont (Reference 100) and Boeing (Reference 101), among others, have evaluated the benefits to be gained by hybrid or mixed fibers in a laminate. The hybrid fiber may be included for specific property improvements. Low-cost glass fibers can be intermixed, in prescribed locations and amounts, possibly in nonmaximum load direction, and may be used to reduce costs.

8. Design for Ease of Fabrication

McDonnell Aircraft has designed the Harrier wing with full consideration given to manufacturing (Reference 102). The concept was to make the design as simple and low risk as possible as a tradeoff with weight optimization. The aim was to minimize layup time and risk of loss of the part during fabrication.

9. Material Control

The contribution of material control in reducing manufacturing costs must not be overlooked. Material control includes storage conditions, packaging, and special handling procedures. Useful, efficient handling of material is essential to minimize waste. Lockheed Missiles presents insights and suggestions on the proper control of composite materials (Reference 34).

10. Out of Autoclave

An autoclave (or press) of sufficient size to produce a full-size DC-9 wing represents a major investment in equipment. Many development activities are underway to eliminate the need for the autoclave.

A. The pultrusion and roll-forming processes do not yet produce final cured parts of acceptable quality. Final cure under heat and pressure is required.

B. Vacuum Curing Resin

Resin systems are under development that can be used with graphite and processed with heat and vacuum bag pressure alone. The usual epoxy systems that perform well with autoclave pressure will cure with air or gas bubbles and high void content when cured under a vacuum bag. TRW and several prepreg and resin suppliers are working on a solution. Northrup (Reference 92) reports major cost savings and only marginal lowering of strength properties with a vacuum bag cure.

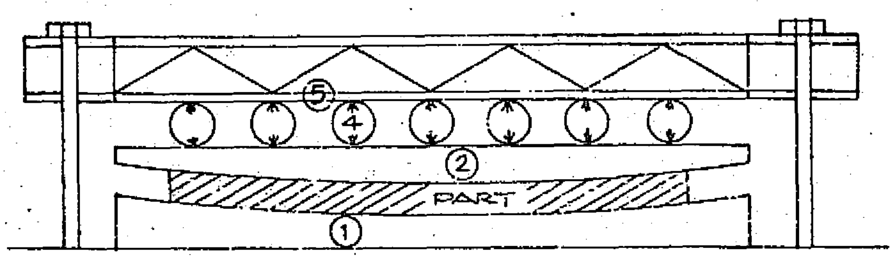
C. Trapped Rubber Molding

The trapped rubber molding process has been partially developed at Lockheed and at Douglas. Douglas has manufactured a series DC-10 upper rudder by this process (Reference 94). The concept uses the high coefficient of expansion of silicone rubber, enclosed in a rigid metal box so that with heat, the box contains the silicone and develops pressures suitable for compacting a composite part during cure. Heat is supplied by an oven or by electrical heaters imbedded in the tool. This process is presented with further detail in Section 7.

D. Inflatable Fire Hose Pressure

The fire hose pressure molding concept supplies pressure to the back side of a mold surface much like a large hydraulic press. A typical installation would have a rigid lower surface tool in a fixed location ① . An upper surface tool would match-mold the upper surface of the part ② . The back side of the tool would be virtually flat. A series of inflatable fire hoses ④ would then be located between the upper tool and a rigid upper surface (perhaps a rigid frame truss construction) ⑤ .

The hoses are inflated to produce the desired effective pressure on the part. Movement is fairly restricted and requires proper design to allow placement of the part to be cured in the tool. Only small potential energy is stored in the volume of the hoses, there is no pressure bag to break, and the support tool is not expensive to build compared to an autoclave. This concept has been used occasionally throughout the industry with the most notable known success at Bell Helicopter.



Real energy savings potential exists for a tool ① and ② that contains an electrically heated surface plate backed by a rigid insulation structure. In this case, only the tool surface and the part itself would be heated to the high curing temperatures. If the tool surface were made of graphite/epoxy, of the same fiber pattern as the part to be cured, and if the backing insulation had low thermal coefficient of expansion and low thermal conductivity, the resulting tool should be very stable to thermal distortion due to curing temperatures. Close dimensional control should then be possible for the fabricated parts.

11. Low-Cost Tooling

Tooling for composite manufacture for long wing-type structures is presented in Section 7. The following text presents a discussion of two unique low-cost tooling concepts used to produce large composite parts that may prove useful for the wing program.

A. Graphite/Epoxy Sandwich Tool

A large tool was fabricated by General Dynamics and used to manufacture the F-5 fuselage midsection (Reference 105). A plaster master mold was built using standard plaster technology. A graphite/epoxy skin was cured under vacuum bag pressure and low [60°C (149°F)] heat on the plaster mold. The outer skin was then cocured and bonded to the honeycomb with the same temperature and pressure. After cure, the part was removed from the unharmed plaster, placed in a holding fixture, and postcured to 177°C (350°F). A wet epoxy resin system was catalyzed to partially cure at 60°C (140°F and to completely cure at 177°C (350°F) to obtain 177°C (350°F) elevated strength properties after postcure — without resin softening and without distortion. General Dynamics claim for the concept was:

(1) Low cost for a large, complex shape compared to conventional metal tools,

(2) Excellent thermal stability. The graphite skin was the same fiber pattern as the part to be made later, and no thermal distortion occurred.

(3) The tool was lightweight and did not place severe limits on heat-up capability for a final part autoclave cure cycle.

B. Quartz Fiber/Epoxy Tool

A large tool was fabricated by Boeing for a structural component that was designed similar to the General Dynamics approach except that quartz fiber cloth was used in place of graphite cloth or tape. A general-purpose high-temperature epoxy resin system was used and autoclave curing pressures. The use of quartz was a cost-saving material substitution, and good dimensional stability was also reported (Reference 104).

MANUFACTURING TECHNOLOGY ISSUES

The successes achieved by Douglas Aircraft Company in curing monolithic graphite/epoxy structures such as the DC-10 upper aft rudder have reinforced the manufacturing philosophy that integrally cured structural assemblies will ultimately become the most efficient and cost-effective aircraft construction method. This belief is predicated on the fact that integrally cured assemblies eliminate most mechanical joints, ensure proper fit-up of details, minimize structural weight, and can result in lower manufacturing costs by reducing the time required for assembly. We realize that the technology available today is not sufficient to permit immediate commitments to manufacturing an integrally cured wing. Considerable research effort must be devoted to developing manufacturing techniques to permit large-scale curing of wings. If composite wing structures are to replace aluminum wings on commercial aircraft, then advances in the manufacturing technology must be oriented to take full advantage of the net molding possibilities afforded by composites. Straight replacement of aluminum by graphite/epoxy on a part-for-part approach is not yet cost-effective because of the high cost of the graphite material. Additionally, lower design strain level allowables must be used when graphite parts are joined with mechanical fasteners. Adhesive bonding for primary structural joints is not yet perfected with sufficient

reliability for long-term (20-year) continuous service. From today's viewpoint, the maximum cost/weight benefit from composites will coincide with maximum integral curing of assemblies.

Under present manufacturing capabilities, a composite wing can be produced using a more conventional approach. This approach is to fabricate details such as stiffened wing skins and individual spars and separate ribs or bulkheads, and join the parts with mechanical fasteners. Advantages over aluminum structure may be gained if wing skins are made in one piece, with the stiffening elements incorporated as integral components of the skin. Ribs can be fabricated with lightening holes and stiffeners, and buildups molded in one piece. Metal attachment fittings can be incorporated in the lay-up and cured in position. This concept of wing production is possible today, with a minimum of development effort; however, the total potential benefits, in terms of reduced manufacturing costs, are not as lucrative as those for more integrally cured assemblies.

The manufacturing technology issues discussed in the following paragraphs support the concept of monolithic structure. Descriptions of specialized molding methods, such as inflatable mandrels and trapped rubber, are more applicable to integral curing than to conventional piecemeal construction. The goal presently invisioned for composite wing production is to cure the lower stiffened wing skin with both front and rear spars and to include the 22 ribs in one structure. The upper stiffened skin, molded with the lower structure, is separable after curing by using a Teflon barrier. The removable cover skin provides access to the wing for tool removal and subsequent assembly installations, and still provides perfect matchup of the skin because all parts are cured together. The cover skin, mechanically fastened to the substructure, completes the wing assembly.

Admittedly, this is a rather imaginative approach to wing fabrication, but it is one that we believe should be pursued for more favorable long-term results.

Molding Methods

Autoclave – The autoclave was initially used in the curing process to add pressure to the laminate, allowing better resin infiltration around fibers and developing a more uniform finished product. The current autoclaves have been boosted to meet higher temperature and pressure requirements. This forced heating systems to be converted from the tool to the autoclave atmosphere. As autoclave size increases, the cost of operation rises in proportion to the diameter squared. Also, the cost of the facility rises exponentially with the increase in size.

With this economics-limited situation, manufacturing requires that wing or fuselage structures which are too large to be fabricated in one piece have splices designed for modular construction.

With these larger structures, the risk of loss during cure becomes greater. The large stationary autoclave also dictates the location of the assembly lines and limits the flexibility of the plant layout due to its size. The basic module size developed at Douglas in 1970 for structural component handling was 3.05 meters (10 feet) in diameter and 9.1 meters (30 feet) long. Because of the need for fewer parts and integrated design, along with the weight saving of fewer splices, this size may need modification. The autoclave tooling is operated with balanced pressures in all sides, allowing the use of lightweight mold surfaces just heavy enough to prevent distortion.

Addition of splices to the wing design takes a large penalty in weight, fabrication complexity, and cost. The current design concept involves component sizes for an autoclave with the equipment now available in industry.

Press Molding – Press molding has been the industry standard from the early days of plastics. It is the production approach for most components, but due to its mechanical configuration, the size and shape of the parts limit the use of presses.

The typical press has tie rods at each of the corners, and these limit the size of the tool or part that may be produced. A typical component of graphite/epoxy requires the molding/cure pressure of 0.69 MPa (100 psi) and 177°C (350°F) to properly process the resin. A large aircraft wing panel component measuring 20 by 3 meters (65 by 10 feet) requires a press of nearly 4540 metric tons (5000 tons) and, combined with the platen rigidity necessary, this would cost more than $3 million to acquire. This press would still have problems due to the tooling complexity.

To achieve the goals of this program, the use of presses requires development. The ability to process epoxy resin systems using a hot platen press hinges on the development of the hot strength of the cured resin, to allow the rapid cycling possible on a press. Tooling setups for short runs on a press are efficient only for large-volume parts. Clips are ideal configurations as they are used in many areas.

Press tooling costs 100 to 150 percent more than similar tools used in an autoclave. The dimensional control is better in the press because of the "matched tool concept" where both surfaces are hard-tooled.

An epoxy resin, chopped graphite form, compression molding compound that could achieve 80 percent of the strength of the continuous fiber cured laminate would be press-molded to produce clips and other small components. The labor costs of using bulk molding compounds and the compression molding process are low compared to hand layup and the autoclave cure process. Compression-molded details are usually molded net to shape, including holes, countersinks, cutouts, and other openings.

Inflatable Elastomeric Mandrels — A new development at Douglas has been the use of silicone rubber inflatable mandrels inside a composite structural cavity to provide pneumatically controlled levels of molding pressure. An inflatable mandrel has distinct advantages over other types of molding tools in that it has low thermal mass, is reusable, and can be collapsed and withdrawn through small openings such as access holes in the cured structure.

The inflatables are formed from uncured silicone rubber sheet stock calendered to uniform thickness. Sheet silicone rubber is available with a Dacron woven insert within the sheet which provides a greater tensile strength. This reinforced rubber is generally used around radii where stresses can be high enough to tear the plain rubber sheet. It also acts as an effective doubler over joints and seams.

The uncured rubber is layed up into a female mold, built up to wall thicknesses of 3 mm (0.125 inch) or more, and then vacuum bagged and cured at 177°C (350°F). A pneumatic fitting is embedded and cured in the mandrel wall to introduce pneumatic pressure later during part cure.

When using inflatables to mold straight parts such as blade stiffeners on a skin, the mandrel must be stiffened along its length to ensure straightness of the cured part. Metal inserts and hingeable inserts have both been very effective in maintaining alignment of the mandrel. They also provide a rigid mold surface which can be used for layup.

The present state of inflatable mandrel technology is somewhat unreliable because of leakage through the rubber mandrels. This is the basic problem in extended use of the inflatable molding system. If the mandrel leaks during the cure cycle, the differential pressure between the inside and outside of the tool is lost and a poorly molded part results. One concept for avoiding some of the pitfalls mentioned above is to use a very high elongation (1000 percent), thin-wall bladder that is not preshaped. It may be possible to buy mandrels in only a few standard shapes, inflate them to pressurize the part, then peel them out of the part cavity and discard them. Development work is required to find a high-temperature rubber with high elongation properties, like a simple balloon.

The inflatable mandrel molding process is a prime candidate for producing integrally cured composite wing structure. Figure 3-13 illustrates a potential method of employing inflatables to fit between rib bays in the wing. Each mandrel is formed to mold the wing skin stiffeners and the ribs simultaneously. The mandrels are pressurized via a common pneumatic line to provide equal pressure. As shown, the cover skin is cured with the spars and ribs, but separated from them by a barrier film. The cover skin is reattached to the wing structure later after tooling is removed and piping installations are completed.

Trapped Rubber Processing — Trapped rubber processing, using the thermal expansion method, has proven to be a viable means for producing one-piece cocured composite box structures such as the DC-10 upper aft rudder. Tooling is currently being initiated to produce a DC-10 nose landing gear door by this method. The process is still in its infancy and there are many unknowns and variables. As these problems surface and are resolved, the process will become a valuable manufacturing technique and could prove beneficial in the fabrication of a composite wing box.

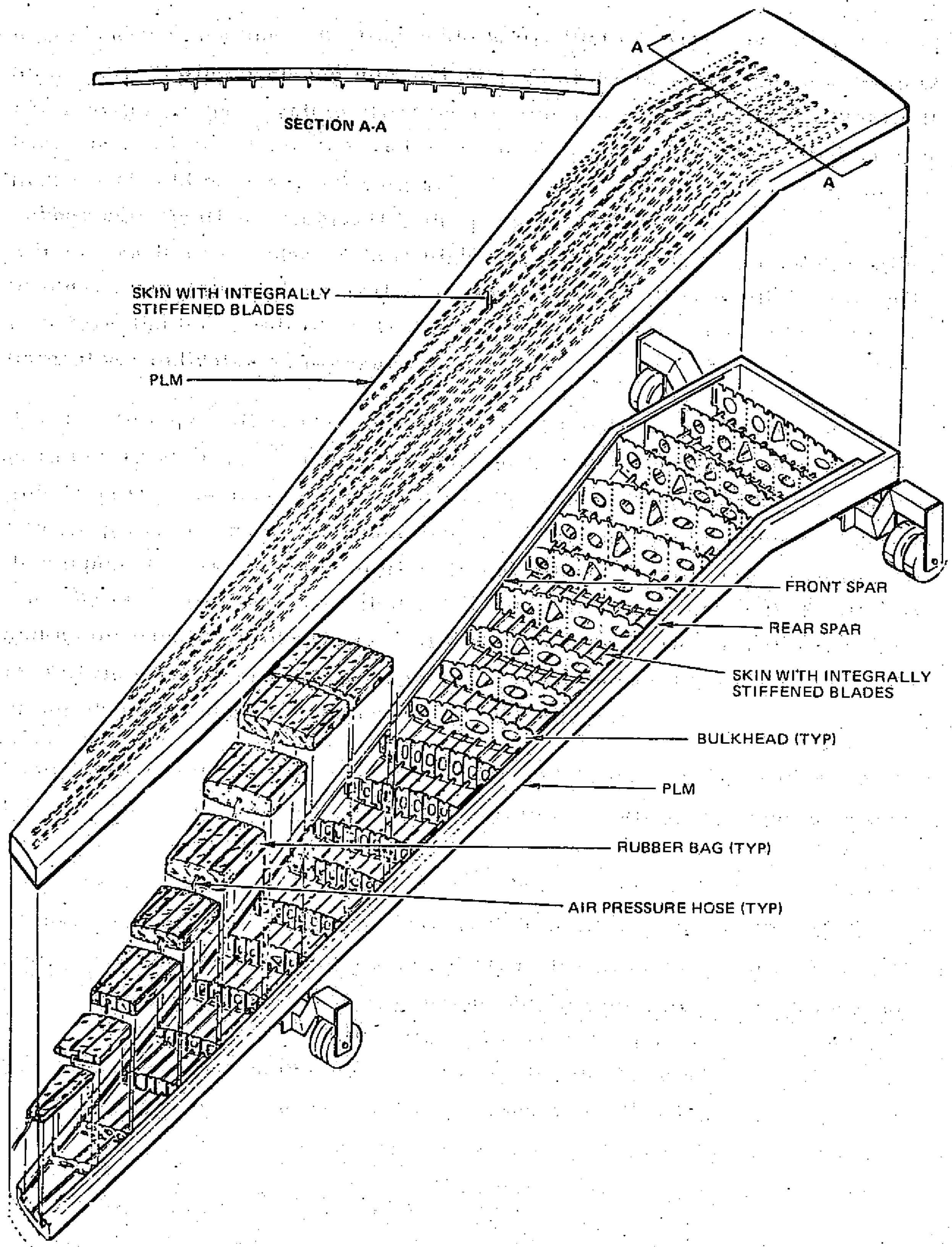


FIGURE 3-13. DC-9 COMPOSITE WING CONCEPT

Trapped rubber processing is predicated on the thermal expansion properties of elastomeric-type materials, primarily room temperature vulcanizing (RTV) silicone rubbers. The term applies to a molding process for composite materials in which precast silicone rubber is placed within a closed (but not airtight) cavity and allowed to thermally expand against the composite's surface supported by the walls of the vessel. This generates pressure internally rather than externally as by standard means (vacuum bag, autoclave, or platen press). The process can be modified and used in conjunction with vacuum bag and autoclave methods where high pressures [in excess of 0.69 MPa (100 psi)] are not required.

Among the advantages over conventional methods, trapped rubber processing:

- Eliminates the use of an autoclave and its costly operation
- Eliminates standard vacuum bag techniques and the problems associated with leakage and bag failures during the cure cycle
- Eliminates costly bagging materials which can be used only once
- Requires only a standard air-circulating oven (or the tool can be self-contained with internal and/or external heaters)
- Eliminates the risk of losing a part during a cure cycle due to faulty bagging or pressure loss
- Permits reuse of mandrels without danger of pressure loss.

The disadvantages of the process are:

- Extremely heavy and bulky tooling (dependent upon part configuration)
- Slower heating rates due to tooling mass
- Part configuration must be adaptable to this process.

The composite wing box could be a choice candidate for production by trapped rubber processing as it lends itself to an open box configuration; i.e., the integral curing of ribs, bulkheads, and spars to a lower skin with a removable upper skin, or a cured egg-crate construction with separately attached upper and lower skins. Tooling could be limited to: (1) a closed system with an air-circulating oven and internal heaters as required to enhance the heating rate, or (2) an open system utilizing a vacuum bag and autoclave. Within the time span allotted for design and development, problems associated with the process should be resolved; namely, gap control, pressure control, and rubber stabilization. Currently, the process is dependent upon RTV-type silicone rubbers because of their high thermal expansion rates, but evaluation of other materials such as Teflon or Nomex, should they exhibit more efficient pressure and temperature control and better tool indexing, could make them desirable candidates as pressure media.

The DC-10 upper aft rudder investigation will address pressure-sensitive parameters of trapped rubber. Sufficient technical information will be available to permit an accurate definition of the rubber configuration in order to control pressure levels as a function of rubber and cavity volumes. Additional development work should be initiated to eliminate the precision gap requirement by incorporating controlled voids within the cast rubber that will collapse and automatically limit pressure at a chosen value. This capability would permit the casting of rubber net into the tool volume without careful adjustment of gap through accurate fabrication of a dummy part.

Combination Molding Process — Inflatable and Trapped Rubber Process — The trapped rubber process has been used advantageously in sections less than 30 cm (12 inches) deep, with access holes to permit removal of rubber tooling after cure. Control of molding pressure is accomplished by correct sizing of rubber and cavity volume ratios during tooling construction. For deep sections that would require large volumes of rubber, the pressure control is more difficult and the rubber provides a considerable thermal mass which increases the curing cycle time. Additionally, silicone rubber is expensive and adds considerably to the tooling costs. Minimizing the volume of trapped rubber within the tool can reduce rubber costs, energy utilization, and cycle times to make a more efficient process. Thus, efficiency can be enhanced by combining the trapped rubber with an inflatable mandrel to occupy a high percentage of the tool volume and provide molding pressure.

A Company-sponsored program is underway to incorporate trapped rubber molding in conjunction with an inflatable mandrel to fabricate a wing box section. The test box will determine the effectiveness of curing components of a wing, including blade-stiffened skins, with the cover skin separable from the box after cure. A removable cover is necessary for extracting tooling, for access to install subsystems such as fuel piping, and for inspection of the cured composite. The cover is mechanically attached to the front and rear spars and to separately cured chordwise ribs which are installed after the box is cured.

Figure 3-14 is a schematic view of the box end showing the trapped rubber strips used to generate horizontal molding pressures on the sides of the blade stiffeners. Metal mandrels between the blades provide straight molding surfaces and also act as layup and densifying forms for the graphite/epoxy tape used on the box. Each mandrel was densified to compact the layup close to final thickness, enabling fitup into the tool.

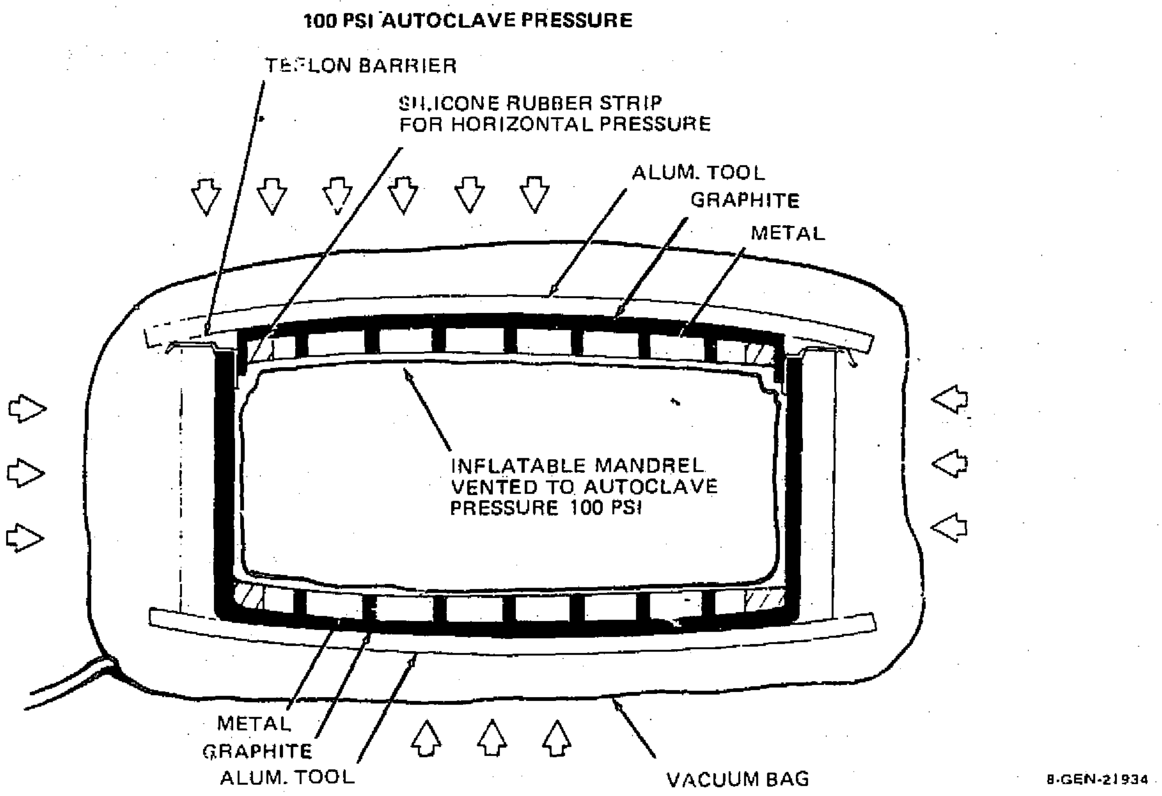


FIGURE 3-14. COCURED WING BOX MOLDING METHOD

Pressure to cure the skin areas was provided by an inflatable silicone rubber bag formed within the tool on dummy parts prior to layup. By venting the bag to autoclave pressure, control of molding pressure normal to the bag surface can be achieved by the normal autoclave pressurizing system. The net differential pressure level across the tool wall is zero.

The female box tool was purposely designed as a low-cost, low-mass tool which would be vacuum-bagged and would not be required to withstand high differential pressures between the interior and exterior surfaces. Aluminum was selected as the tooling material because of its availability, high thermal conductivity, and machinability.

The first part has been successfully cured using the inflatable mandrel formed from uncured silicone rubber with a seam along the upper edge where the bag contacts the sidewall.

This concept of the combination process has a high potential for curing wing sections with ribs, and is considered as a viable method for future assemblies. More development work must be conducted to find a reliable inflatable mandrel and to simplify layup of composite into the tool.

In order to integrally cure a wing structure and avoid the fastener installation costs, weight penalties, and part fitup problems associated with joining separately molded composite parts, the combination process requires a development program to address the following areas:

- Construction of reliable inflatable mandrels
- Verification of the combination process on a subscale part
- Manufacturing cost data to determine effectiveness of cocure molding concept.

Tool Development

Low-cost Tooling Concepts — Tooling required for molding large structural parts can be massive, causing high thermal lag, and is expensive to construct. Simplified tooling approaches are desirable in conjunction with combination molding processes to avoid large pressure differentials across molding tool walls.

Conventional composite tooling generally consists of steel molding surfaces machined to contour on multiaxis N/C milling machines, then supported by complex truss work. The costs associated with producing such tools are high, and they can be justified only where many parts will be produced. As the composite wing technology program will construct only three wing box sections, the use of low-cost tooling would be very advantageous.

Low-cost tools can be fabricated with plane surfaces and containment of molding elements in picture frames, Figure 3-15. A flat or curved sheet of aluminum or titanium defines the contour of a stiffened skin. Integral blade stiffeners are cured by trapped rubber pressure and the autoclave generates curing pressure in orthogonal directions. Such simple tooling has been used to construct test panels for the NASA Composite Specimen Program (NAS1-12675).

Another approach to low-cost tooling is the use of castable materials which are readily swept into final form by the use of templates. Castable ceramics have been demonstrated to be effective in curing graphite and Kevlar epoxy parts. By sweeping the mold to shape, little or no machining is necessary to generate compound curved surfaces. The ceramic tool can be integrally heated for curing the composites and the mold surface can be permanently coated with a release material to ensure that parts will not bind to the mold. The thermal coefficient of expansion of ceramics is on the order of 0.56×10^{-6} cm/cm/°C (1×10^{-6} in./in./°F), which closely matches that of graphite/epoxy.

Tooling has been fabricated using graphite/epoxy layups with flex-core stiffening. This approach provides excellent thermal match between the mold and part, but is costly to construct because of the manual layup of the graphite.

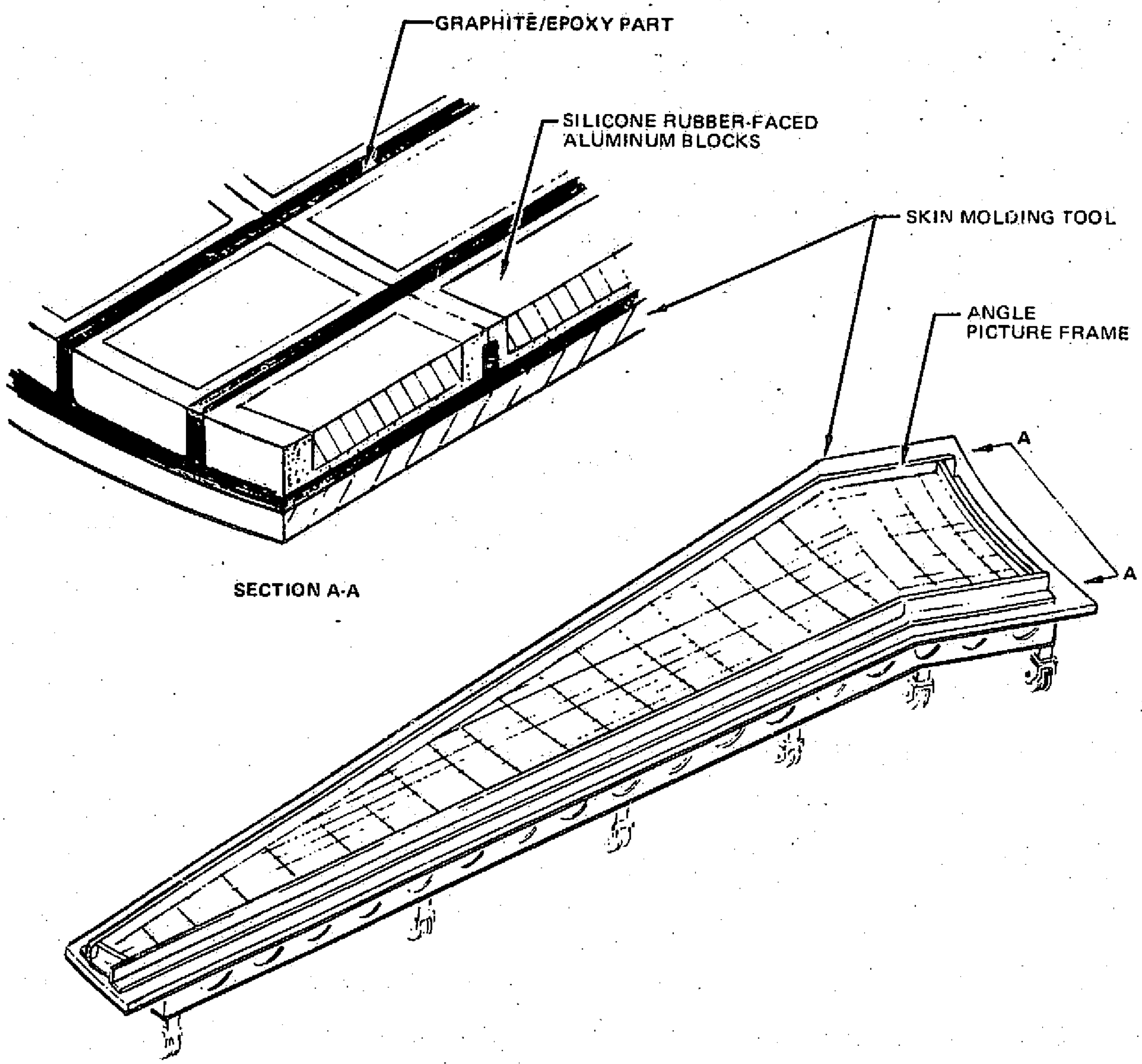


FIGURE 3-15. LOW-COST TOOLING

Block graphite has been used as a tooling material where thermal expansion must be very low. However, the nature of graphite block is such that the material must be machined in a special facility to minimize dust and contamination of equipment. Douglas has jobbed out all block tooling because of the machining problems. A special sealing coat must be applied to the contact surfaces to prevent adherence of the part to the mold. We have used graphite block tooling to only a limited degree.

The application of simple tooling is predicated upon the configuration of the final part. A very simple aluminum mandrel formed from bar stock can be used back-to-back to produce H-beams. By designing the web surface to remain flat, the relative motion between the part and tool during cooldown does not load the part. In fact, the aluminum tool will shrink away from the graphite part, enhancing release.

Very accurate aluminum tooling has been used to mold integral hat-stiffened panels of high quality representing wing skin thicknesses and compression loads. Aluminum is readily machinable with good surface finish. Cast aluminum plate has been tested as tooling material but lacks the smooth finish possible with 6061-T6 alloy.

The greatest potential for low-cost tooling is in the castable tool where machining can be avoided. Splashes taken from plaster master parts can be used to build tools either by back-casting or by making female molds from materials such as fiberglass. Tooling can be developed for the contoured wing skins by using several castable materials to determine tool costs, dimensional control of the final part, and longevity of the tool after repeated curing cycles.

Stationary Tooling – Three approaches for the in-line stationary tool that could be incorporated into the production line for a single-use dedicated facility are: (1) autoclave, (2) hot-oil bladder, and (3) thermal expansion rubber.

Autoclave Curing – This is the conventional approach but involves using multiple- and single-purpose tools in an autoclave-type pressure vessel. The small-volume autoclaves would be adjacent to the assembly line. The autoclave design would conform to the shape of the part with self-contained heating and cooling capabilities.

The disadvantage of this type of curing is the bagging and potential for leakage. Better systems need to be developed for assembly of the prepreg on the tool, applying the bleeder materials, sealing the vacuum bag film to the tool, and testing and loading prior to cure, as well as the need for development of the curing mechanism itself.

For high-cycle curing, a better heat transfer is offered by the use of a built-in rubber blanket tool that circulates heated oil instead of static gas for pressurization. This puts the heat source in direct contact with the composite material for maximum efficiency. After satisfying the cooling requirements, the oil temperature is easily reduced by circulating it through exchangers, and then circulating to cool the tool and part. The feasible pump pressure available for circulating hot oil is about 0.35 MPa (50 psi).

This technique has limitations, especially with the silicone rubber bags because of their low strength at elevated temperature. With circulating hot oil, further hazards may arise from leaks. Aside from damage to the curing parts, personnel safety must also be addressed.

The borderline pressure of 0.35 MPa (50 psi) on the system can also create cure problems. Normal process specifications require 0.69 MPa (100 psi) for curing structural epoxies. Pressure below the amount specified may result in a deficiency in sharp corners or in deep draw areas.

Thermal expansion molding has rapidly developed in the last five years. The unique characteristics of certain silicone rubber compounds permitted the development of a tooling material called RTV. RTV rubber expands under heating, a characteristic which can be converted into pressure for curing composites; it contracts on cooling, causing a tooling mandrel to shrink and ease its removal from inside a cured part.

The major program using this approach is the DC-10 composite upper aft rudder. This is a structural part over 3.5 meters (12 feet) high, molded in one piece using a steel tool to the outer mold line. The prepreg graphite/epoxy broad goods and tape are layed up, densified, and formed. These details are then stored in the freezer until assembly. The details are loaded into a mold for final cocuring and bonding in one operation. The tool is bolted together; the assembly is rolled into the oven. Using external convective heating with the internal heat through cartridge heaters, the rubber expands and the part is cured.

Large-Scale Tools — Composite molding tools for 15-meter (50-foot) parts have not been designed or tested to determine where problem areas might occur. Wing contours employ compound curvatures and twist along the wing axis. Dimensional control of the contour is very important and extraneous warpage of a wing skin during cure would be unacceptable. Conventional tooling experience is limited to relatively small parts where thermal effects cannot be adequately extrapolated to full-sized wing sections.

Tooling materials have historically stressed longevity over economy or thermal properties. A serious consideration for large molding tools is the thermal mass which directly influences the heating rate during cure. Tool designers customarily using massive plates to construct large tools must reexamine the traditional materials and masses surrounding the composite material to promote uniform heating at the prescribed rates.

One of the basic problems associated with maintaining dimensional control of a cured wing is the thermal expansion of the tool and the compressive stresses induced on the part during cooldown. When expandable rubber curing is used, this problem can be severe enough to fracture parts. Methods of relieving compressive forces on the part by the tool could be used to avoid breaking good parts. The problem may be attacked by the selection of low-expansion tooling materials to match the thermal expansion characteristics of the graphite/epoxy material or by designing relaxation mechanisms into the tool.

Many aircraft tools use welding to fabricate strong, rigid joints from steel stock shapes. The residual stresses induced by the welding operation could produce undesirable warping of the tool when subjected to the $177^{\circ}C$ ($350^{\circ}F$) temperatures associated with curing graphite/epoxy. Stress-relieving may be necessary at progressive stages of tool construction. Tool designs may require symmetry of sections or balanced masses about a specific axis to avoid warping.

The present programs funded by NASA will not deal with tools on a size typical of the composite wing. Answers to these problems will be found by constructing large tools, measuring the thermal and fabrication effects, and compensating the design of tooling to minimize these problems on the production tooling.

PROGRAMMATIC ISSUES

The programmatic issues shown in Figure 3-2 are not classified as key issues because they will be demonstrated in the course of constructing and testing flightworthy hardware. An airframe manufacturer will not commit to production of composite wings until a high degree of confidence exists that low-weight flightworthy structure can be produced on schedule for predictable costs. The data and experience needed can only be gained by the design, manufacture, and test of a flightworthy composite wing box which contains a range of design features representative of those to be encountered in a new commercial transport wing.

Data Base

A comprehensive design data base is essential in the development and qualification of any new wing design. A data base for in-house commercial transport aircraft has accumulated over the years from development and qualification testing of the DC-6/7, DC-8, DC-9, and DC-10 aircraft, supplemented by many tests on Douglas-built military aircraft and other data from NASA and various industry and government sources. The data base includes test data correlated with analytical prediction, data on the development of analytical methods, and a library of technical manuals and standards.

The data base for a new model is composed of all applicable data supplemented by additional test and technology development to account for new design features, size effects, and recent regulatory changes. The expansion of an existing data base represents a modest investment in time and cost when compared to the generation of a totally new data base.

The existing data base for aluminum wing structure cannot be used for a composite wing box design and a new data base must be generated. Data accrued from the NASA ACEE and other government-funded composite structures programs and from in-house composite development projects will form the nucleus of the new data base. This base must be supplemented by data representative of the composite wing box size, materials, layup patterns thicknesses, processes, manufacturing methods, structural design features, and other characteristics.

The DC-10 program can be used to illustrate the application and expansion of a data base. (See Figure 3-16.) Results of more than 2000 DC-8/DC-9 fatigue and fail-safe tests were used to evaluate the preliminary design configuration of the DC-10. During the DC-10 development and detail design phases, 300 bow-tie specimens and 140 wing subcomponent specimens were added to establish the lg stress levels for the DC-10 wing structure. Many other development and verification tests were conducted for static strength, corrosion protection, and the like. Full-scale static proof load tests were conducted on the second flight article, and the fourth production airframe was dedicated for full-scale fatigue and fail-safe verification tests.

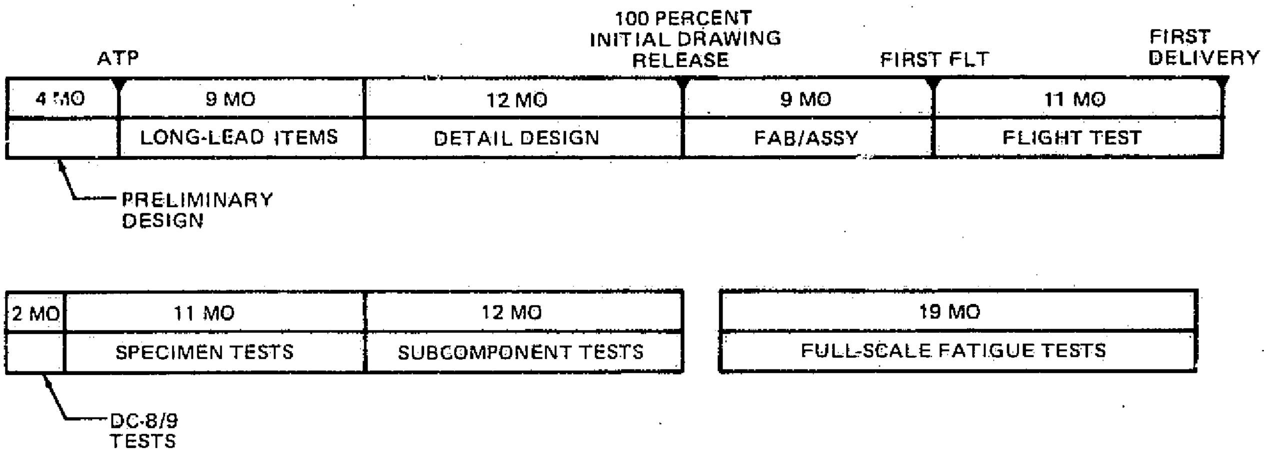


FIGURE 3-16. DC-10 STRUCTURAL DESIGN AND TEST SCHEDULE

The existence of an applicable data base at the onset of a new composite wing box production aircraft program serves the following purposes:

1. To provide evidence to management and airlines that a structurally reliable composite wing box can be produced.

2. To have data immediately available for the design synthesis phase. Otherwise, schedules must be extended to account for time to conduct development tests.

Further, the cost of producing a totally new data base could adversely affect a production commitment to CWB structure.

Weight Estimates

The decision to utilize a composite wing box design in a new production aircraft is highly dependent upon the weight savings that are obtainable Projected weight savings based upon an optimistic conceptual design may be compromised as the design synthesis progresses and design parameters are

C.2

introduced which adversely affect the optimum weight. More data are required to establish that predicted weight savings are valid.

- Durability and damage tolerance criteria may require that lower design strain levels be established than those on which weight predictions were based.
- Increased accessibility requirements for manufacturing, inspection, and in-service maintenance and repair may impose structure inefficiencies greater than those reflected in the conceptual design.
- Fabrication and assembly methods required to produce cost-effective structure may require weight tradeoffs.
- Less weight-efficient layup and ply orientation patterns may be required to avoid geometric distortions for proper fit.
- Additional lightning protection features may be required to preclude in-tank arcing and other damage.
- The addition of metal parts may be required to improve strength in a direction normal to the ply layup.

The negative case has been presented. In the same sense, the conceptual design could be conservative and additional weight savings may be attainable to enhance the selection of a composite wing box structure.

The weight-estimating techniques used for composite structures are based on techniques proven by correlation of predicted and actual weights of metal and fiberglass parts. Additional correlation is derived from parts produced for the in-house NASA ACEE composite structure programs and the DC-10 upper aft rudder and vertical stabilizer. In-house composite flight evaluation hardware and test articles have also been used to validate the estimating techniques. It has been found that most of the variance of structural weight stems from potential variables in the design rather than in weight-estimating techniques.

Schedules

Delivery schedules for new aircraft are highly competitive and the composite wing box structures must be available to meet the assembly schedules.

Figure 3-17 presents a schedule which is considered typical for a new aircraft to be developed in the mid-1980s. This schedule shows that the wing must be completed and ready to be joined to the fuselage structure 19 months before the first aircraft delivery date.

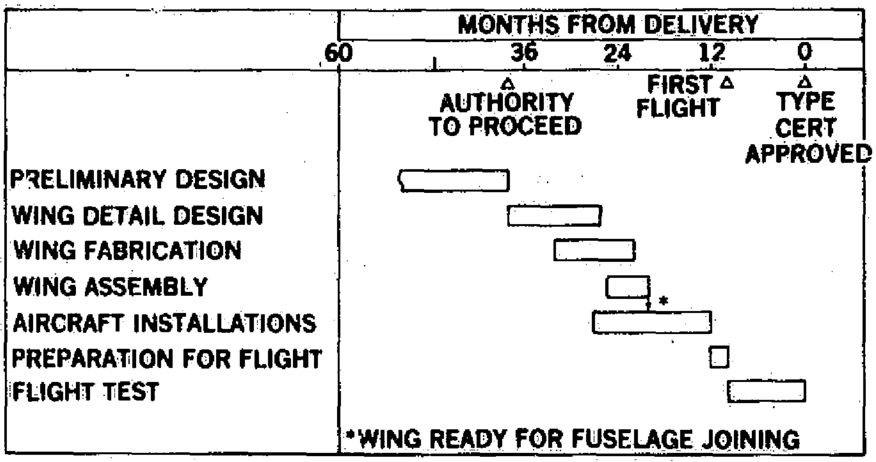


FIGURE 3-17. TYPICAL SCHEDULE DATA FOR A NEW PRODUCTION AIRPLANE

The composite wing box structure poses two schedule issues:

1. A low risk must be associated with composite wing box production schedules. If the wing is not ready on schedule, the aircraft will be delivered late. Contracts for the delivery of new airplanes usually include late delivery clauses to the effect that the airline must be recompensed to offset the added expense of providing alternate aircraft and for the loss of revenue which the newer model would have generated.

2. The new aircraft go-ahead decision is made after airplanes have been sold and delivery schedules are firm. At this point, only 19 months remain to complete the detail design and produce the first composite wing box. To reach this point, management must have committed to a composite wing box preliminary design several years earlier. Unless a very high confidence level exists, one would also need to carry forward a conventional

wing design with the implied added development costs to prevent any delays in meeting the schedule.

Cost Estimates

In making a commitment to utilize composite primary wing structure in a new production aircraft, the manufacturer will want the increase in benefits of the venture to be proportionate to the increase in risk. The reliability of cost predictions is a fundamental risk element.

At present, it would be inappropriate to attempt anything more than a rough-order-magnitude cost estimate for composite primary wing structure for civil transport aircraft. Design data and manufacturing data are available from other programs which, coupled with a preliminary design, could provide a cost model. However, for a large wing structure, too many factors must be considered which interact to affect the cost of the final product.

Cost estimates to support a firm production program commitment must be made on the basis of very early preliminary design information. As the design progresses, it may happen that design features will have to be incorporated which prevent the use of the intended cost-efficient concepts.

The synthesis of the preliminary design for composite structure involves a knowledge of the manufacturing methods, inspection methods, and a need for accessibility for manufacturing, inspection, and in-service maintenance and repair. Weight optimization and structural integrity must also be considered. The facilities and equipment that will be available must be compatible with the production rates and may dictate a less efficient manufacturing approach. Engineering may specify ply stacking sequences for structural integrity which exclude cost-effective automated methods. Advanced but unproven low-cost manufacturing methods must be weighted against the risk of delays and higher rejection rates.

When all of these and other factors are considered, there is little justification to place much credence in cost predictions based on a preliminary design with no historical wing data to support the predictions.

A composite wing technology program which includes options for the design, manufacture, and test of full-scale composite primary wing structure for civil transport is essential before the manufacturer can place any reliance on his predicted costs.

Experience

A large, experienced staff will be required for the composite wing box production program. Management must be assured that capable personnel are available to create a minimum-weight, low-cost design, and to produce high-quality, certifiable structure on schedule.

The new airplane's first delivery schedule limits the time available to expand and train the composite wing box staff. An experienced cadre must exist to train and supervise new personnel, to develop structural and manufacturing technology for long-lead-time tasks, and to provide technical expertise to develop a technically acceptable preliminary design.

Capability must be established in the structural design team, materials and process engineers, manufacturing engineers, quality assurance personnel, and production fabrication and assembly workers to assure a balanced and coordinated composite wing box production program.

The experience base that will be provided by NASA ACEE composite structure programs, in-house programs, and other government-funded composite programs must be further expanded before the composite wing box production program is started.

SECTION 4
WING SELECTION

A baseline wing design is a prerequisite to the conceptual design of the composite wing box structure for the following reasons: (1) for weight saving, cost, schedule, and trade study comparison, (2) to define the scope of the development program, and (3) to determine facilities and equipment required.

Five aircraft wings were evaluated as prospective candidates for the composite wing technology program. Parameters considered included the following:

1. The vehicle should be a commercial transport aircraft with a range of design features to adequately demonstrate wing technology.

2. The wing should be a reasonable size to be cost-effective.

3. It should have the geometry, structural loads, environmental exposure, utilization rates, and FAA certification requirements typical for a future production aircraft.

4. Design data such as criteria, external loads, loft lines, and interface requirements must be readily available.

5. An aircraft must be available for a composite wing flight evaluation program. This implies certification by the FAA and subsequent revenue operation by a commercial airline.

These factors imply that the candidate wing options are limited to civil transport aircraft manufactured by the development plan contractor and currently in airline service, or at least far enough into development to ensure that design data are available and that an airplane will eventually be available for flight evaluation.

Accordingly, the five candidate wing options shown in Figure 4-1 were considered during the wing study:

1. The wing of the model C-15 STOL aircraft for the United State Air Force.

Two prototype YC-15 aircraft have been built and flown. Since the study started, the Air Force has discontinued plans for C-15 production and NASA now owns the two prototype aircraft. This airplane wing has the right size for the development plan with adequate structural features for resolution of key issues. However, the airplane is not a civil transport and would not be suitable for in-service evaluation.

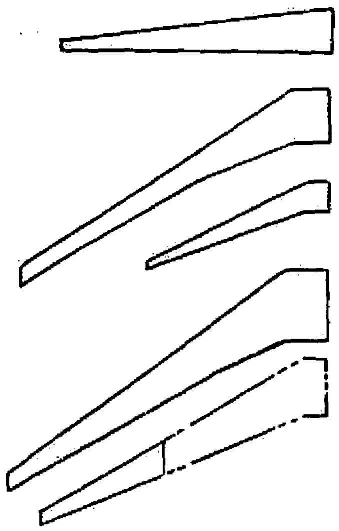
2. A DC-X-200 is currently in advanced design. Wing design data are available. The wing is a high-aspect-ratio supercritical wing which is probably representative of a 1985-1990 production aircraft and might be a good choice if a flight evaluation phase were not required in the development plan.

3. The DC-10 wing satisfies all requirements except that it is too large for cost and schedule factors. The design of the wing box does not lend itself to a spliced outer composite wing except outboard of the fuel tanks. This outboard section does not sufficiently represent inboard wing design features to address all the key issues (fuel tank wing-to-fuselage joining, main landing gear attach heavy structure, etc.).

4. Many DC-8 aircraft are still flying and design data are readily available. The full-span wing box is too large for an economical program, but the outboard wing has a design joint to the inboard wing. The size of the outboard DC-8 wing is ideal, but the objection mentioned for the DC-10 outboard wing applies equally to the DC-8 outboard wing: it is not representative enough to address all the key issues.

5. The DC-9 wing has the best attributes for the composite wing structural development plan. It is small enough so that the full wing can be used, data are available, many aircraft are in commercial service, and the wing design is representative. Figure 4-2 shows the similarity of the main box geometry of the DC-9 wing to the next-generation DC-X-200 supercritical wing box geometry.

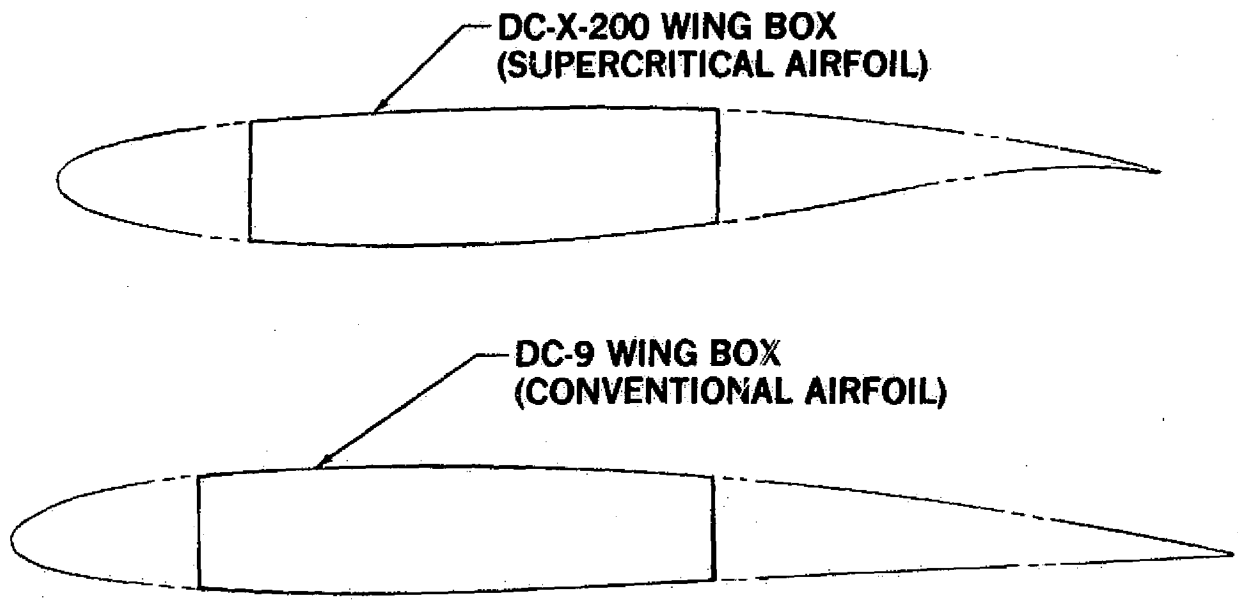
Of the five wing options considered (Figure 4-1), the DC-9 aircraft easily outranked the other candidates on the basis of the parameters. There have been many different DC-9 models delivered to airlines. More than 320 model DC-9-32s have been delivered and the same wing is used on several



MODEL	MAX PANEL LOAD K/IN.	AREA SQ FT	COMMERCIAL AIRPLANE	COMPLEXITY	1985-1990 PRODUCTION
C15	12	461	NO	MAXIMUM (4 ENGINES)	POSSIBLE
DC-X-200	12	510	YES	MAXIMUM (2 ENGINES)	YES
DC-9	18	209	YES	MINIMAL (TOTAL PROBLEM RANGE)	YES
DC-10	26	941	YES	MAXIMUM (2 ENGINES)	YES
DC-8	4 (14)	186	YES	INADEQUATE (LOCAL PROBLEM AREA)	NO

8-GEN-21833

FIGURE 4-1. WING OPTIONS



7-GEN-25763

FIGURE 4-2. COMPARISON OF WING BOX GEOMETRY other models, including freighter versions, the Air Force C-9A/VC-9C, and the Navy C-9B aircraft. The DC-9-32 airplane is still in production, and the vehicle will be available for flight evaluation in the mid-1980s. For these reasons, the DC-9-32 was selected for the conceptual design and development plan. The DC-9-32 wing structural arrangement is shown in Figure 4-3.

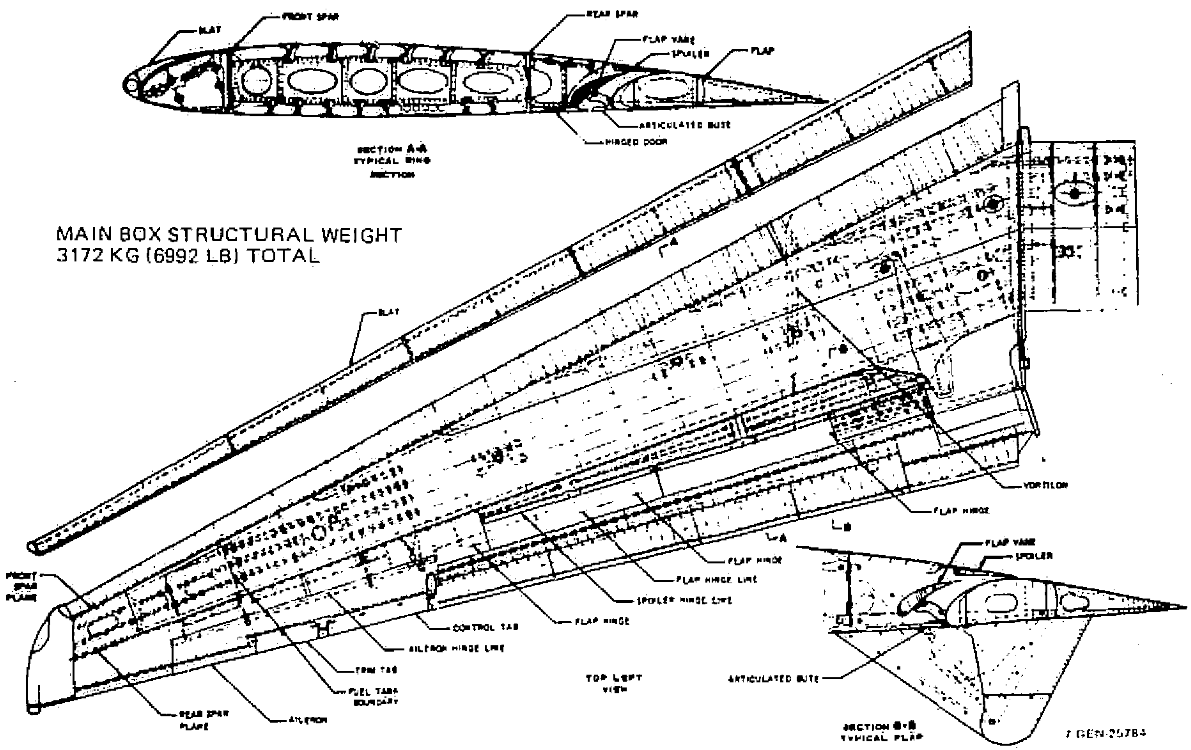


FIGURE 4-3. DC-9-32 WING STRUCTURAL ARRANGEMENT

SECTION 5
PROGRAM OPTIONS

Five program options have been conceived to provide a realistic basis for a contractual technology development effort that will resolve the issues which have been assessed for composite wing box structure. The options are shown in Figure 5-1. All options feature a DC-9-32 aircraft as the baseline configuration around which consistent, well-balanced, and comprehensive plans are formulated. In composing the candidate options, the aim was to establish a set of alternatives which could be used to compare cost, technical risk, and schedule. The timeliness of completing the technology development in relation to future aircraft programs is an important selection factor.

PROGRAM PHASING

Program phasing has been implemented for better management control. Progress can be monitored within each phase and each phase can be separately funded, reviewed, and evaluated for effectiveness in achieving program objectives.

Phase I — Preliminary Design

Phase I of the development plan includes a technology development and the preliminary design of a DC-9-32 composite wing box structure. All technologies which affect the preliminary design are exercised in this first phase. Structural design criteria must be established, trade studies conducted, and design layouts completed. A comprehensive structural test program must be conducted (see Figure 5-2) to provide data to support the choice of structural arrangement and design features.

Unlike metal structures, the manufacturing methods to be used must be decided during initial design. Extensive manufacturing technology development is included to ensure that a practical design is developed. See Figure 5-3. This holds true for access for inspection and repair, lightning protection design features, and other technical and economic design paramters. A synthesis of these preliminary design parameters is presented in Figure 5-4.

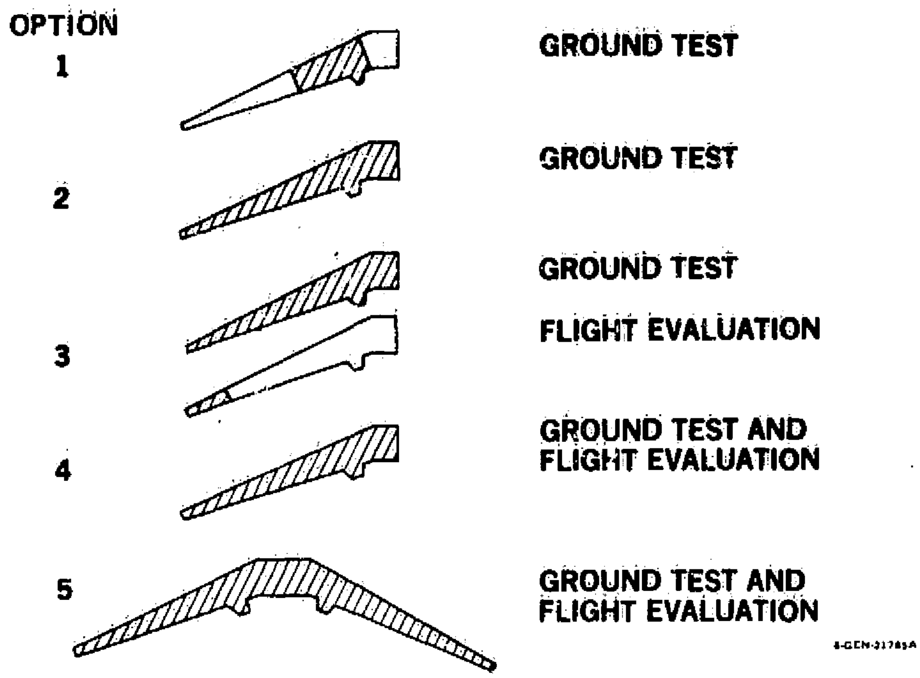


FIGURE 5-1. PROGRAM OPTIONS EVALUATED

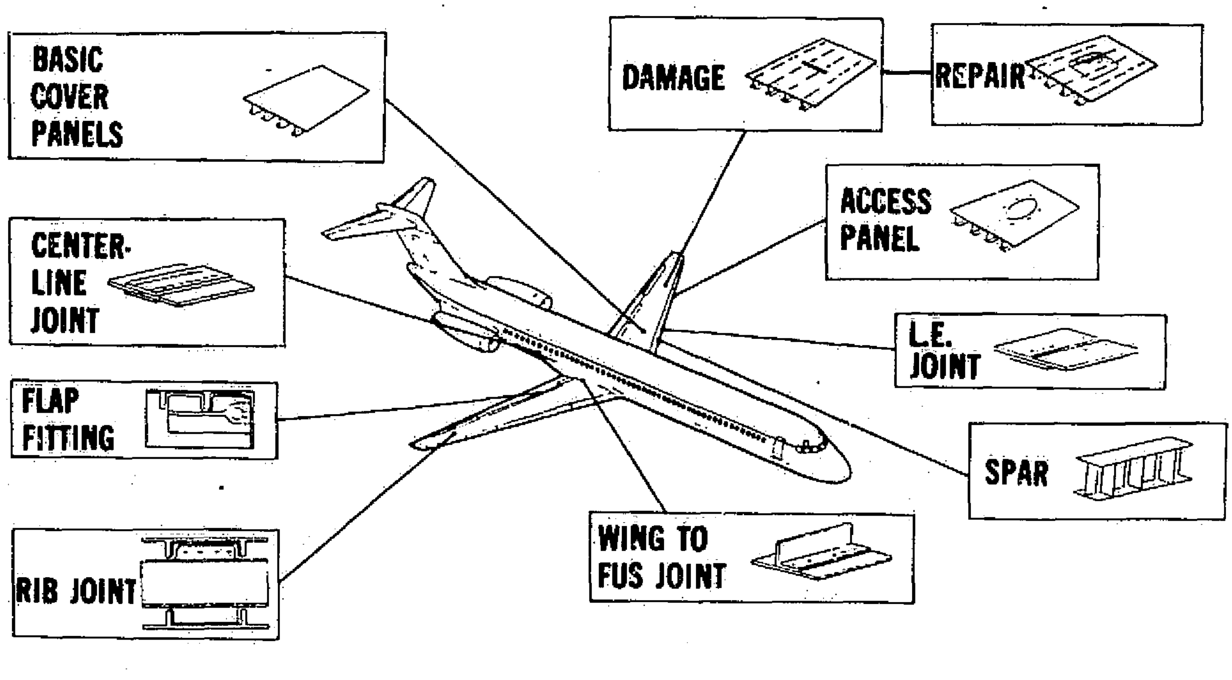


FIGURE 5-2. STRUCTURAL DEVELOPMENT TESTING – ALL OPTIONS

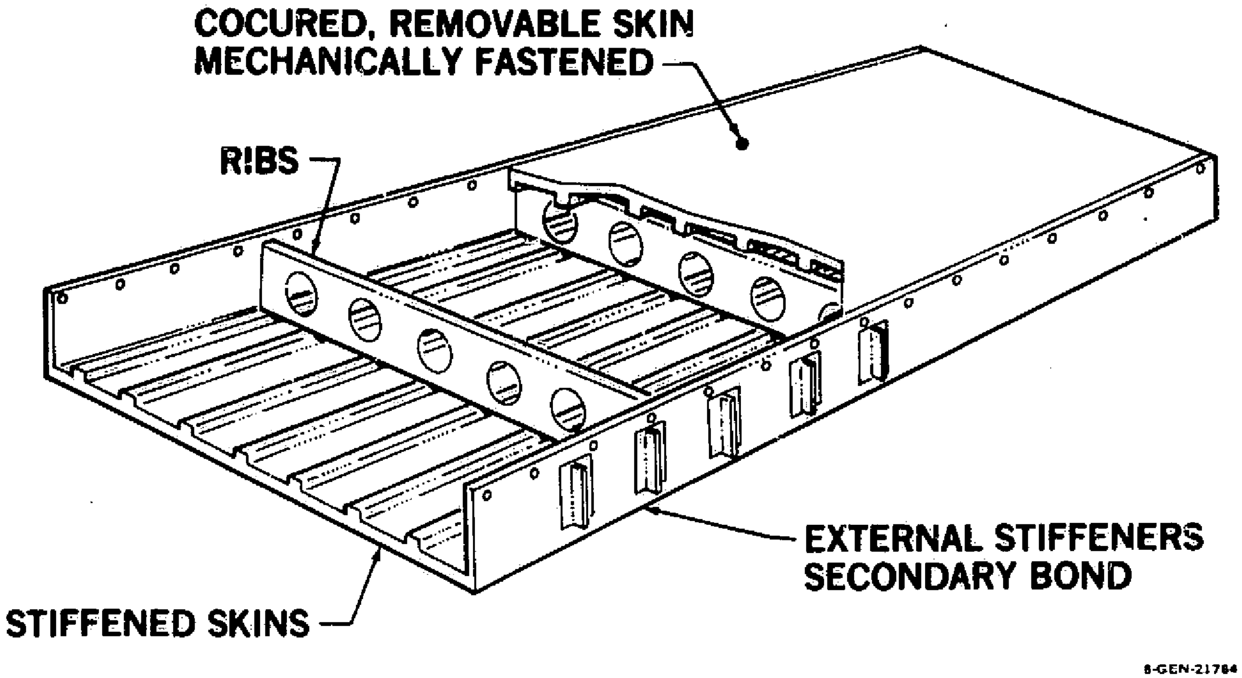


FIGURE 5-3. MANUFACTURING TECHNOLOGY — ALL OPTIONS

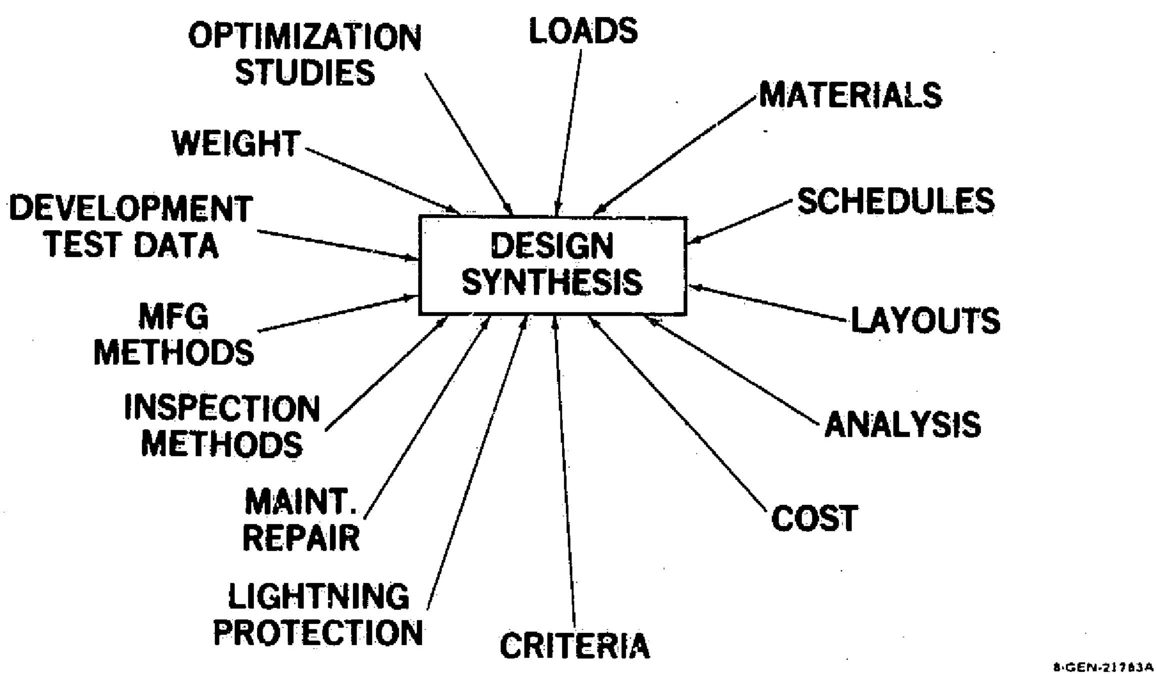


FIGURE 5-4. PRELIMINARY DESIGN — ALL OPTIONS

Phase II – Detail Design

Phase II converts the preliminary design layouts into a detail design from which drawings are produced in order to manufacture hardware. Strength analyses are performed to ensure structural integrity and final criteria. Loads analysis, strength analysis, and weights analysis reports are prepared and submitted to the FAA for substantiation of compliance with applicable regulations. Verification tests are conducted on specimens representative of the final design to provide allowable strength data and to validate manufacturing processes before starting component manufacture. For a Phase II summary, see Figure 5-5.

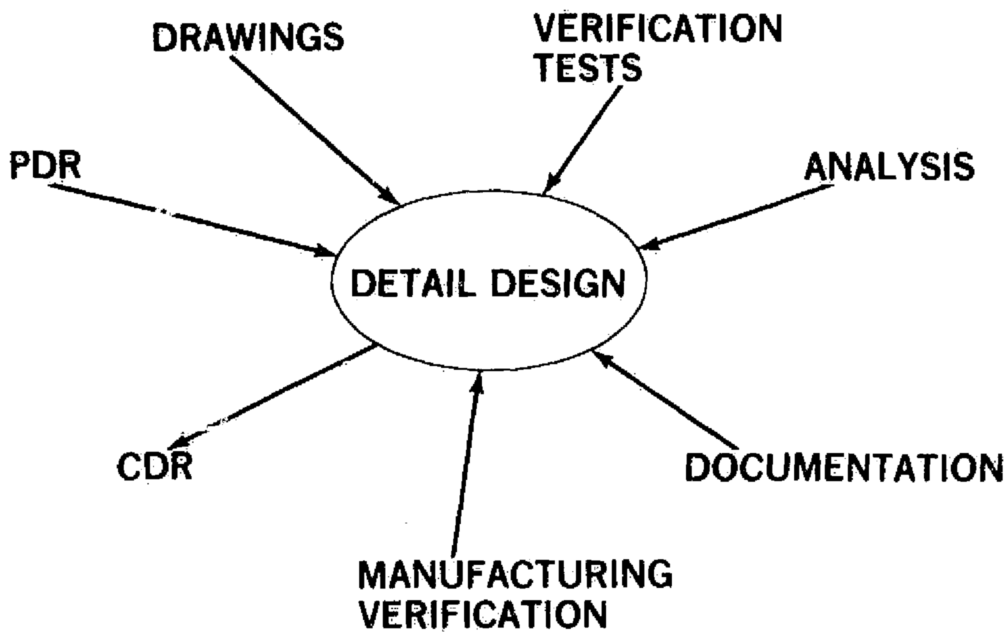


FIGURE 5-5. PHASE II DETAIL DESIGN

Phase III – Manufacturing

Phase III covers the manufacture of components for full-scale ground testing. Tool design and tool fabrication, manufacturing planning, specifications, process controls, and quality assurance are all included in the manufacturing phase. See Figure 5-6.

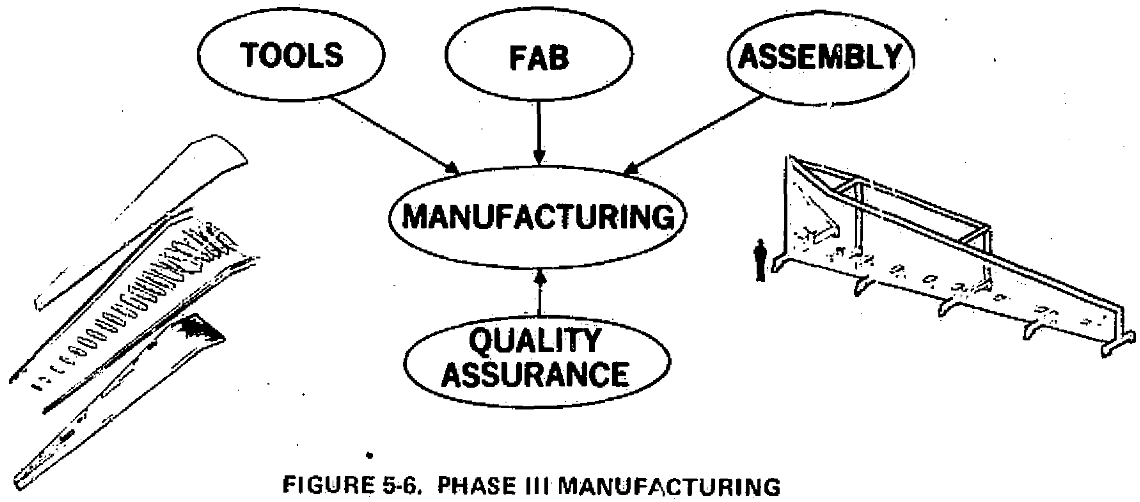


FIGURE 5-6. PHASE III MANUFACTURING

Phase IV — Verification Testing

Phase IV covers the full-scale static, fatigue and damage tolerance, and crashworthiness testing. Tasks in this phase include preparation of the composite wing box to accept test loading fixtures, test planning, fabrication and setup of test hardware, the actual testing, data acquisition, and preparation of test reports. The objectives of Phase IV are to validate the structural integrity of the final product in accordance with FAR Part 25 requirements. The test articles will also be utilized for crashworthiness and repair of major damage tests. See Figure 5-7.

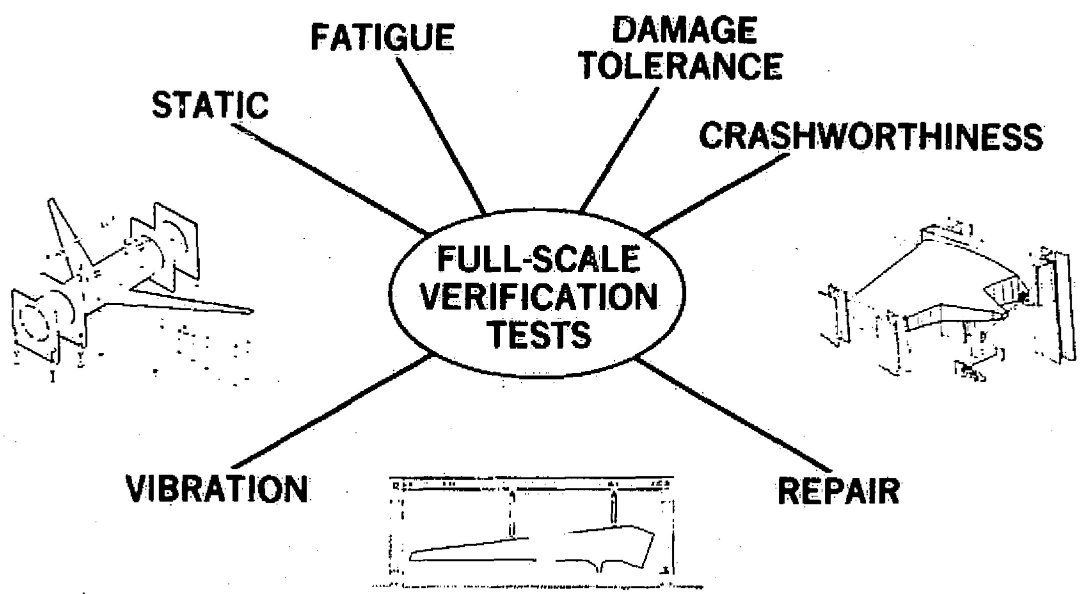


FIGURE 5-7. PHASE IV FULL-SCALE VERIFICATION TESTS

Phase V — Flight Development

Phase V covers flight development, and starts with the fabrication of a composite wing box for installation on an aircraft. The acquisition of a DC-9-32 aircraft and the modification of the aircraft to accept the composite wing box are Phase V tasks. FAA and manufacturer-required ground tests of the aircraft in the flight configuration must be completed and data submitted to the FAA for type inspection approval (TIA) before FAA pilots will fly on the aircraft to witness and approve flight tests required for type certification (TC). After flight tests are completed, the aircraft is refurbished to remove test equipment and configured for delivery to a commercial airline. See Figure 5-8.

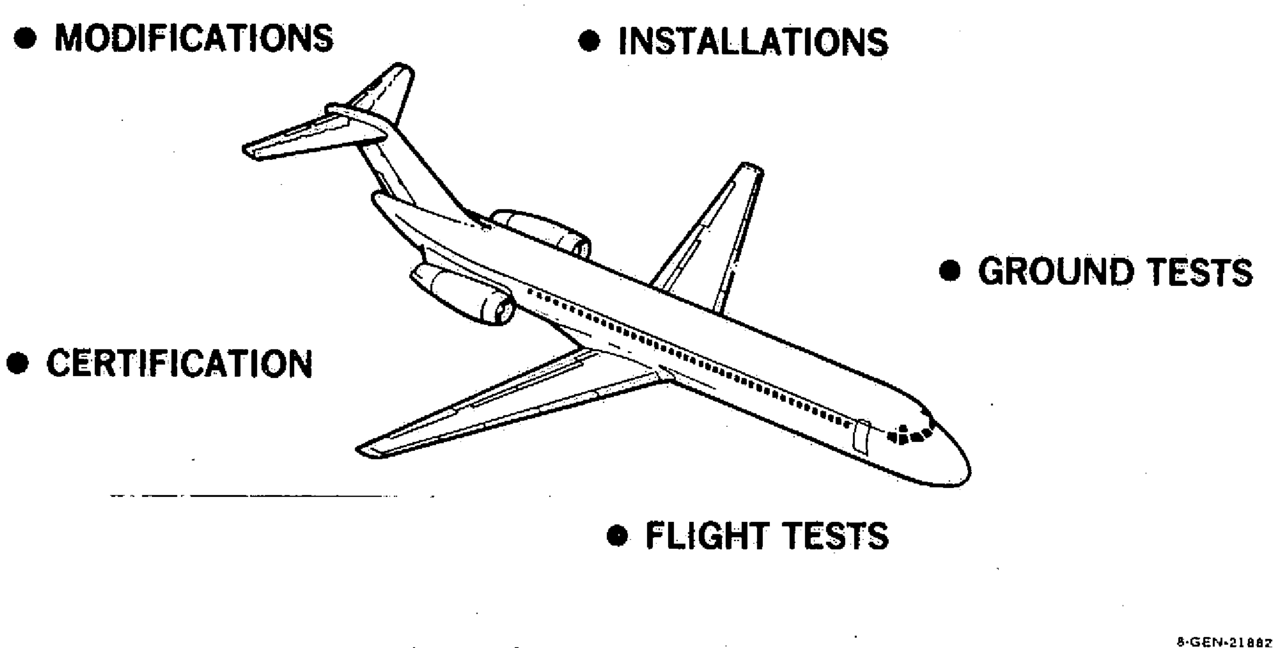


FIGURE 5-8. PHASE V FLIGHT DEVELOPMENT

Phase VI — In-Service Evaluation

Phase VI covers monitoring of the aircraft with a composite wing box after the airplane is delivered to an airline operator for normal revenue operations. The composite wing box will be inspected at intervals and by methods in accordance with an FAA-approved plan. The manufacturer will monitor the inspection program, provide special repair procedures as required, evaluate durability in the civil transport environment, and submit periodic reports to NASA and the FAA to document the structural performance of the composite wing box. See Figure 5-9.

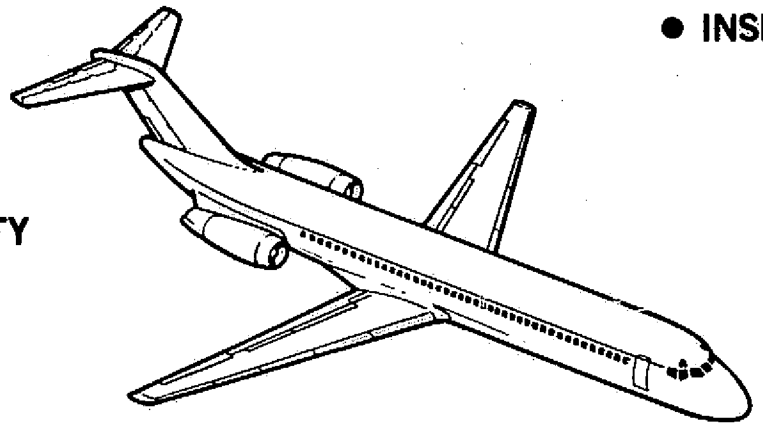


FIGURE 5-9. PHASE VI FLIGHT EVALUATION

DESCRIPTION OF PROGRAM OPTIONS

Table 5-1 presents a summary of all five program options considered. It was considered essential to include provisions in all five options to (1) acquire technology and data, (2) gain design experience, (3) manufacture representative wing hardware, and (4) interface with the FAA to demonstrate the certification procedures for composite structures. The variation between options is therefore limited to the size and quantity of hardware to be manufactured, the amount of testing to be accomplished, and whether a flight evaluation program should be included.

The quality of the technology and data is influenced by how closely the development program is representative of a new aircraft program. Options which do not produce flight hardware can feature structural arrangements and design concepts more ideally suited to composite structures. Options which specify a flight evaluation program are constrained by the need for composite hardware to interface with existing subsystems and adjoining metal structures as discussed in Section 6. Compromises must be made which reduce the cost/weight benefits of the composite wing. These compromises are offset by the touch of realism added to a program which produces flightworthy hardware.

TABLE 5-1
PROGRAM OPTION SUMMARY

PHASE	PROGRAM FEATURES	PROGRAM OPTION 1	2	3	4	5
PHASE I PRELIMINARY DESIGN	DESIGN SYNTHESIS	X	X	X	X	X
	LAYOUTS	X	X	X	X	X
	STRUCTURAL DEVELOPMENT TESTS	X	X	X	X	X
	MANUFACTURING TECHNOLOGY DEV	X	X	X	X	X
	REPAIR TECHNOLOGY	X	X	X	X	X
PHASE II DETAIL DESIGN	DETAIL DESIGN	X	X	X	X	X
	SUBCOMPONENT VERIFICATION TESTS	X	X	X	X	X
	MANUFACTURING VERIFICATION	X	X	X	X	X
PHASE III MANUFACTURING	MAJOR SUBCOMPONENT	(4)				
	SEMISPAN WING BOX		(3)	(3)	(3)	(1)
	FULL-SPAN WING BOX				–	(2)
	CRASHWORTHINESS TEST BOX		(1)	(1)	(1)	(1)
PHASE IV FULL-SCALE VERIFICATION TESTS	STATIC	X	X	X	X	X
	FATIGUE AND DAMAGE TOLERANCE	X	X	X	X	X
	CRASHWORTHINESS	X	X	X	X	X
	REPAIR OF MAJOR DAMAGE	X	X	X	X	X
	VIBRATION	X	X	X	X	X
PHASE V FLIGHT DEVELOPMENT	SEMISPAN WING BOX				(1)	
	FULL-SPAN WING BOX					(1)
	OUTBOARD WING BOX			(1)		
PHASE VI SERVICE EVALUATION	SEMISPAN WING BOX				X	
	FULL-SPAN WING BOX					X
	OUTBOARD WING BOX			X		

X INDICATES OPTION CONTAINS PROVISIONS FOR THE NOTED FEATURE
( ) INDICATES NUMBER OF UNITS TO BE MANUFACTURED

Knowing that the final product will eventually be used in revenue service will imbue the same attitudes in the members of the development program team that exist in those assoicated with a production program. In the same sense, greater confidence in the technology and data produced from a flight program can be expected from the commercial transport aircraft community.

Option 1 — Subcomponent Wing Development

Program Option 1 is composed of unconstrained Phase I preliminary design tasks and Phase II detail design and subcomponent verification testing. Phase III full-scale manufacturing technology development and validation are limited to the fabrication of four major subcomponent articles, as shown in Figure 5-10. Phase IV tests do not include a representation of wing-fuselage interaction effects. Analytical substantiation will be provided to the FAA to verify the structural integrity of the composite wing box and the FAA will witness and approve all tests and test data which are used to demonstrate compliance with Federal airworthiness requirements.

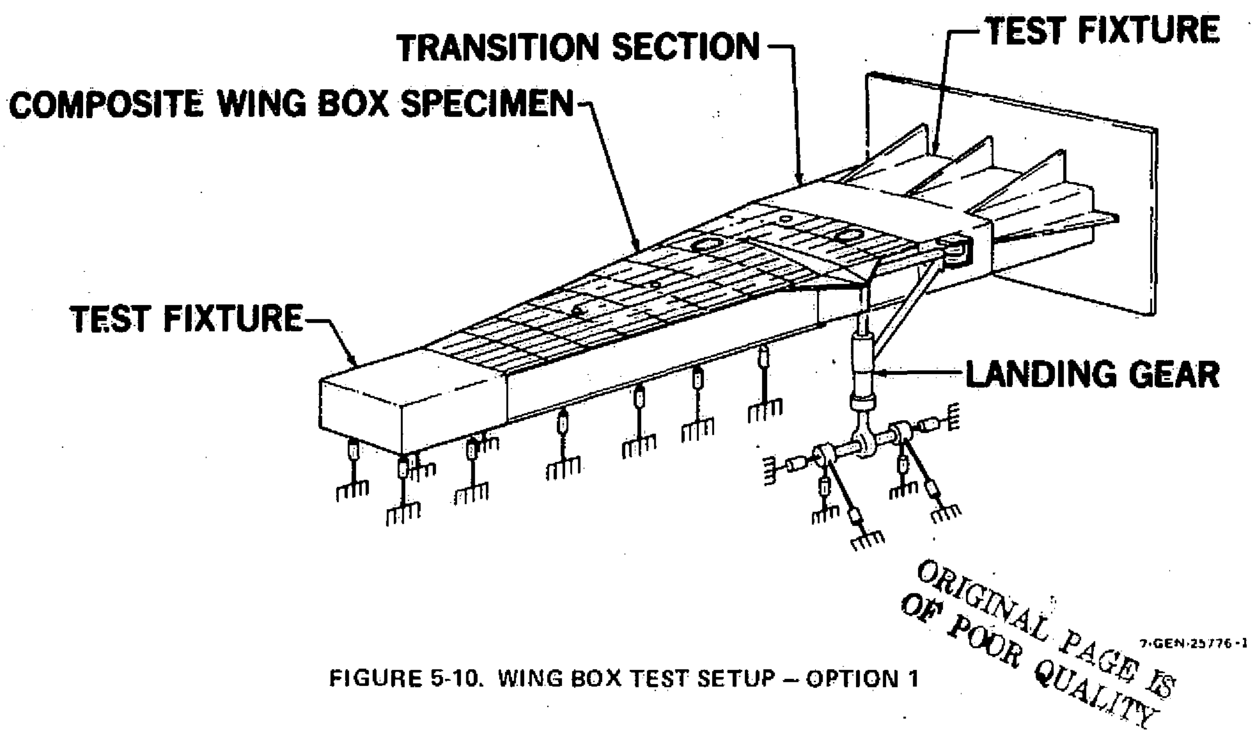


FIGURE 5-10. WING BOX TEST SETUP – OPTION 1

Option 1 is a least-cost program, requires fewer facilities and equipment, and produces data sooner to support a management commitment to a production composite wing box. However, it supplies the least amount of data, does not exercise manufacturing technology to the same extent, does not verify structural integrity to the same extent as other program options which feature full-scale test hardware, and does not provide flight hardware.

Option 2 – Full-Scale Wing Development – Ground Test

Program Option 2 also features an unconstrained engineering design and is different from Option 1 in that the three major subcomponent tests are replaced by the full-scale semispan composite wing box components for static test, fatigue and damage tolerance tests, and a manufacturing development article. The test setups are shown in Figure 5-11. For test purposes, the composite wing box is joined to a production (aluminum) DC-9 wing box, as shown in Figure 5-11. A major subcomponent is added to Phase IV for crashworthiness and repair of major damage verification tests, as shown in Figure 5-12. Option 2 does produce full-scale hardware and can be designed to be more weight efficient than a design which is constrained to meet DC-9-32 criteria and interface requirements.

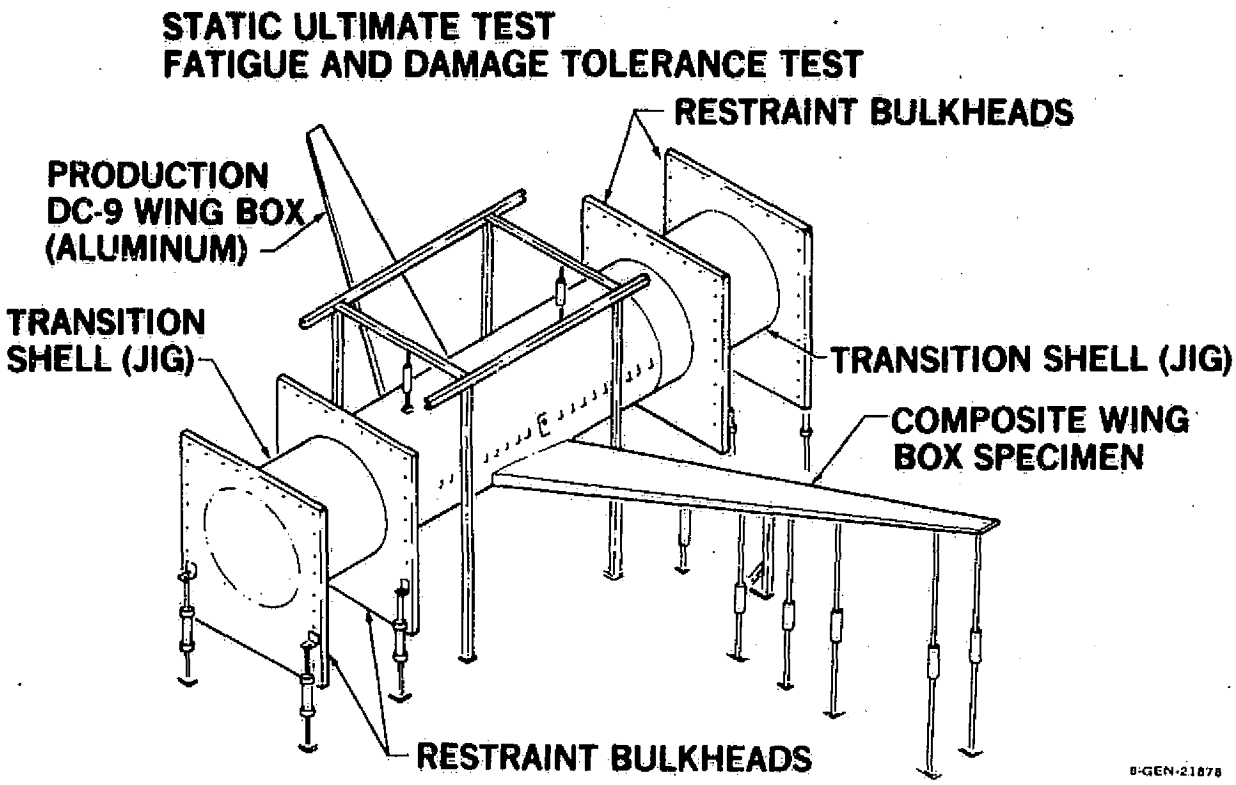


FIGURE 5-11. TYPICAL TEST SETUP – OPTIONS 2, 3, 4, AND 5

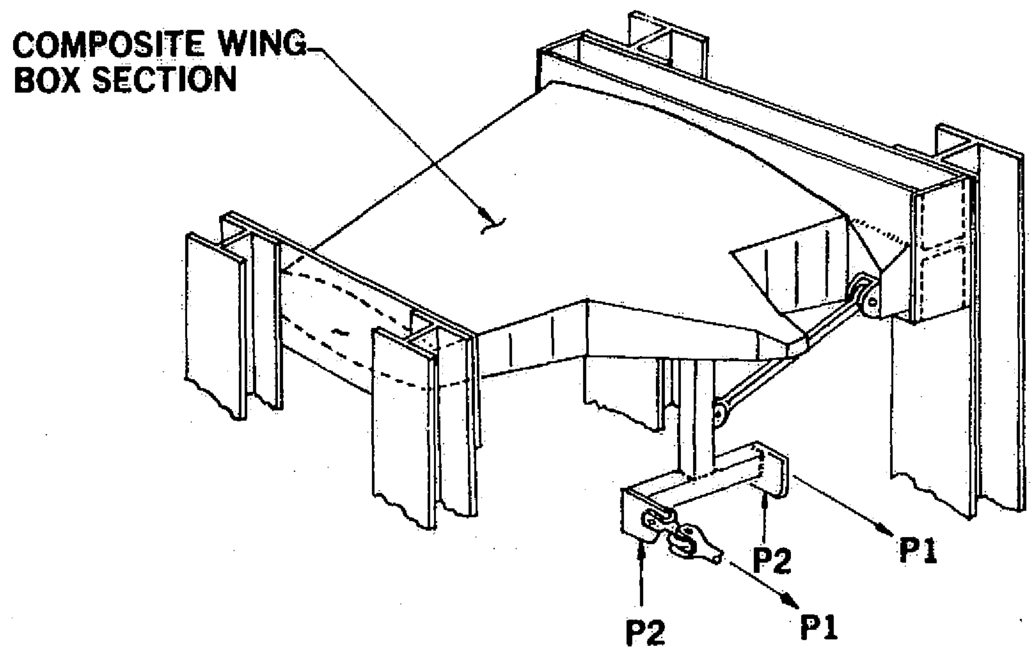


FIGURE 5-12. FULL-SCALE SUBCOMPONENT/CRASHWORTHINESS TEST SETUP

Option 2 is adequate to resolve most of the structural and manufacturing technology and economic issues. The elimination of flight development and in-service evaluation phases offers a significant reduction in program costs as compared to Options 4 and 5.

Option 3 – Full-Scale Wing Development Mini-Flight Evaluation

Program Option 3 is identical to Option 2 except that a composite outer wing box has been added for flight evaluation. The intention is to design the composite wing box without regard for eventual installation of the outer wing box on a DC-9-32 aircraft. After the initial design is completed and used for the manufacture and verification test phases, the composite wing box design would be modified to adapt the composite outer wing to a DC-9-32 aircraft.

Option 3 provides for an unconstrained composite wing box design with a limited flight development and in-service flight evaluation program. The flight component would be outboard of the fuel tank and would not be representative of many of the significant design features of the inboard wing. See Figure 5-13.

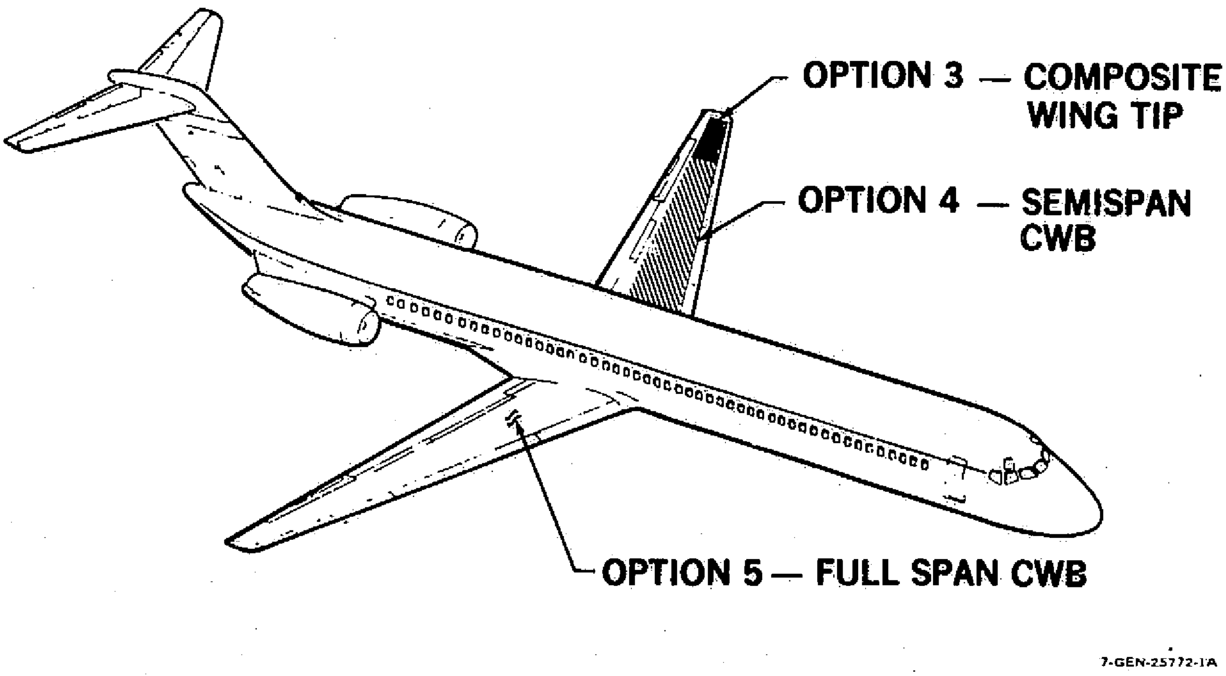


FIGURE 5-13. FLIGHT EVALUATION PROGRAM

Option 4 — Full-Scale Wing Development — Flight Evaluation

Option 4 adds the flight development and in-service evaluation phases to the Option 2 program. Option 2 features an unconstrained design for better weight savings, but the Option 4 design must be constrained to satisfy DC-9-32 criteria and interface requirements. The manufacturing effort is increased to fabricate a semispan composite wing box for flight development. A semispan composite wing box will eliminate the need for the opposite wing tooling and reduce the composite wing box fabrication effort. Since there is a possibility that a favorable management decision can be made without benefit of the flight development or in-service evaluation phases, the program has been organized so that all tasks to produce flight hardware, including engineering redesign of aircraft structure and subsystems (Phase V), can be deferred until data are available from the Phase IV full-scale verification tests.

Option 5 — Full-Scale Wing Development — Full-Span Flight Evaluation

Option 5 is very similar to Option 4. The primary difference in the two programs is that a full-span composite wing box is featured in lieu of a semi-span composite wing box which is spliced at the airplane centerline to an existing aluminum box. Schedules would be the same as for Option 4 and the cost would be increased by the cost to produce right-hand composite wing box tooling and three additional right-hand semispan components, less the cost of two right-hand production aluminum wing box components. The manufacturing development box could still be left-hand only, and the third right-hand box comes with the acquisition of a DC-9-32 aircraft in Phase V.

The full-span composite wing box hardware featured in Option 5 increases program costs over Option 4 with no real gain in technology.

EVALUATION OF ALTERNATIVES

The five program options have been compared to select the option best qualified to form the basis for a contractual technology development effort. The five options were evaluated in terms of relative cost, the time when technology and data would be available, and the extent that the technology gained from each option would fulfill program objectives.

Table 5-2 summarizes the results of the comparison. The range of the variation of relative cost is 47 percent. This can be attributed to the fact that many features are considered essential and are common to all options. The schedule in Table 5-2 refers to the elapsed time from the start of the composite wing technology program to the delivery of the FAA-certified aircraft to an airline for flight evaluation, or to completion of the test program for the option where no flight evaluation is included. The estimated five years to conduct the Phase VI flight evaluation is not included in the table.

TABLE 5-2
PROGRAM OPTIONS
EVALUATION

OPTION	RELATIVE COST	SCHEDULE (YEARS)	TECHNOLOGY GAIN (PERCENT)
1	0.77	5	70
2	0.93	6	85
3	0.95	6	90
4	1.00	6	100
5	1.13	6	100

Program Option 4 offers the best combination of technology gain versus cost with the same availability of technology and data. Option 4 provides the airline with the opportunity for routine inspection, experience, and maintenance. The expected technology gain is considered adequate to impart the level of confidence required for acceptance. Option 5 does not add a significant gain in technology to justify the 13-percent increase in program costs.

From the comparison of the five options, Option 4 is judged to be the most outstanding and will be used as the basis for the formulation of the development plan discussed in Section 7.

SECTION 6
CONCEPTUAL DESIGN

A conceptual design of a DC-9 composite wing box that can replace a metal wing on an airplane was developed for flight evaluation. It forms the basis for the development plan which outlines the design, manufacturing, and testing efforts required for the application of composite materials to the wing of a DC-9 flight article. The design layouts emphasize those aspects of the structure that are unique to composite components and assemblies as well as interfaces with adjoining structure, control surfaces, and systems. This design is also the basis for the weight estimate which indicates the potential for this type of construction.

The intent of the conceptual design was to depict the types, forms, and approximate sizes of structure involved in composite wing design so that associated problems could be foreseen, possible solutions outlined, and the magnitude of the development effort defined. The design is merely offered as representative composite structure. Detail comparative design studies are to be accomplished during the preliminary design phase of the development program itself.

STRUCTURAL DESIGN CRITERIA AND LOADS

The design criteria used for the conceptual design included interface requirements, stiffness, and strength. This approach permitted the definition of the general arrangement and preliminary sizing of structural elements by layout, existing analytical design methods, and available data.

Interface criteria dictate the locations of external support structure. The development program is to be limited to the wing box, but existing slats, aileron, spoilers, flap, landing gear, and fixed leading edge and trailing edge structure must be installed on the box, which is in turn installed on an existing fuselage. This requires that all support structure associated with these installations be located in the same position it now has on the metal wing. Otherwise, the attaching external structure and systems would require redesign.

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Previous in-house studies of composite wing and stabilizer structure indicate that the bending and torsional stiffness provided by the metal box should be maintained in the composite configuration. This ensures that the same load distributions will be imposed on both wing box and attaching structure, such as flaps and slats, and eliminates the need for new loads analysis or redesign of attached structural components. It also ensures the same flutter characteristics and flying qualities. Existing values of bending and torsional stiffness of the DC-9 metal wing are presented in Figure 6-1. These are the values required of the composite wing box concept.

Strength is always a basic criterion in any design. All structural elements including skin panels, spar webs, ribs, attachment fitting installation, and major joints were checked for static strength. Elements not sized for stiffness were designed for strength.

The loads used in this design effort were the same as those determined for the DC-9 metal wing. These include the basic wing bending, shear, torque, and fuel pressure loads shown in Figure 6-2. Skin panel loading is presented in Figure 6-3, and concentrated support reactions are given in Figure 6-4.

CONCEPT SELECTION

The general arrangement selected for the conceptual design is a two-spar, multirib configuration with the spars and ribs in the same locations as in the existing metal design. This selection was based primarily on the interface criteria. The multirib arrangement provides an established approach to the solution of all major problems associated with load paths, interface provisions, and fuel tank requirements. An effort was made to establish viable concepts for composite application to the extent required for development plan definition.

The basic structure consists of blade-stiffened skin panels and shear-resistant, stiffened laminate spar and rib webs. This choice was based on stiffness criteria and previous Company-sponsored studies of the application of graphite/epoxy structure to the DC-9 wing box. These studies included the evaluation of a number of different skin panel concepts. A multirib arrangement was assumed, and J-stiffened, blade-stiffened, hat-stiffened,

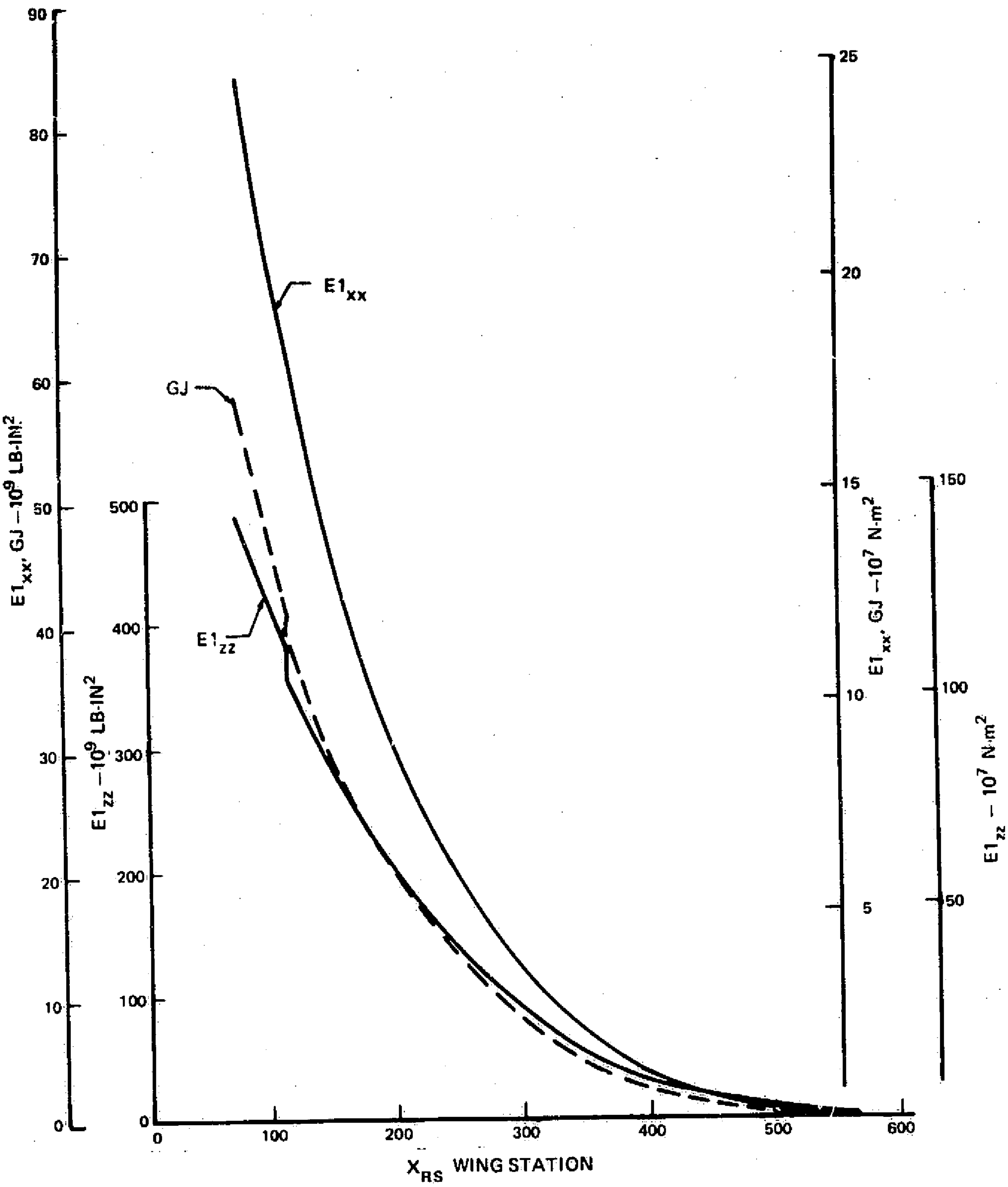


FIGURE 6-1. DC-9 METAL WING BOX STIFFNESS

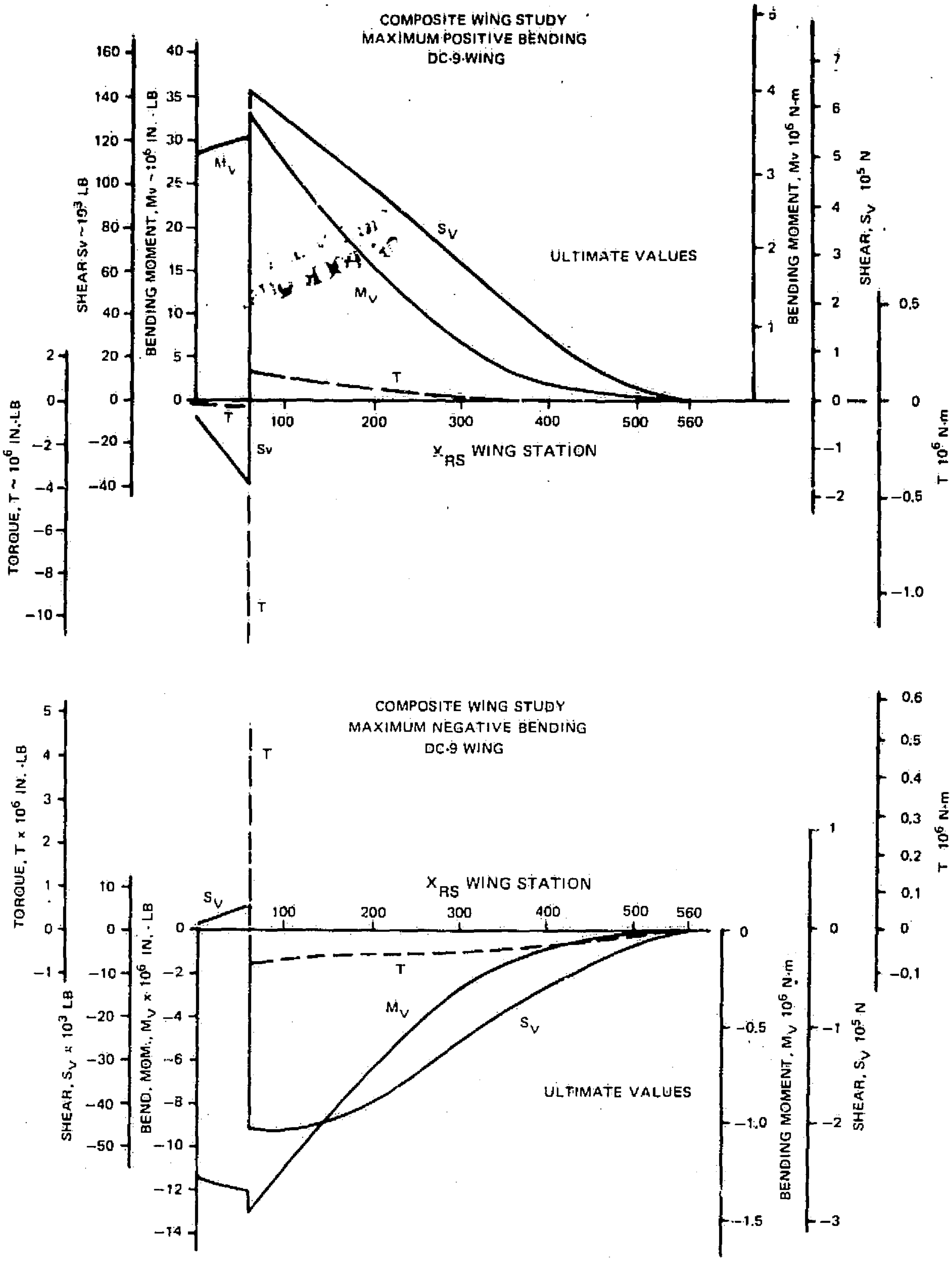
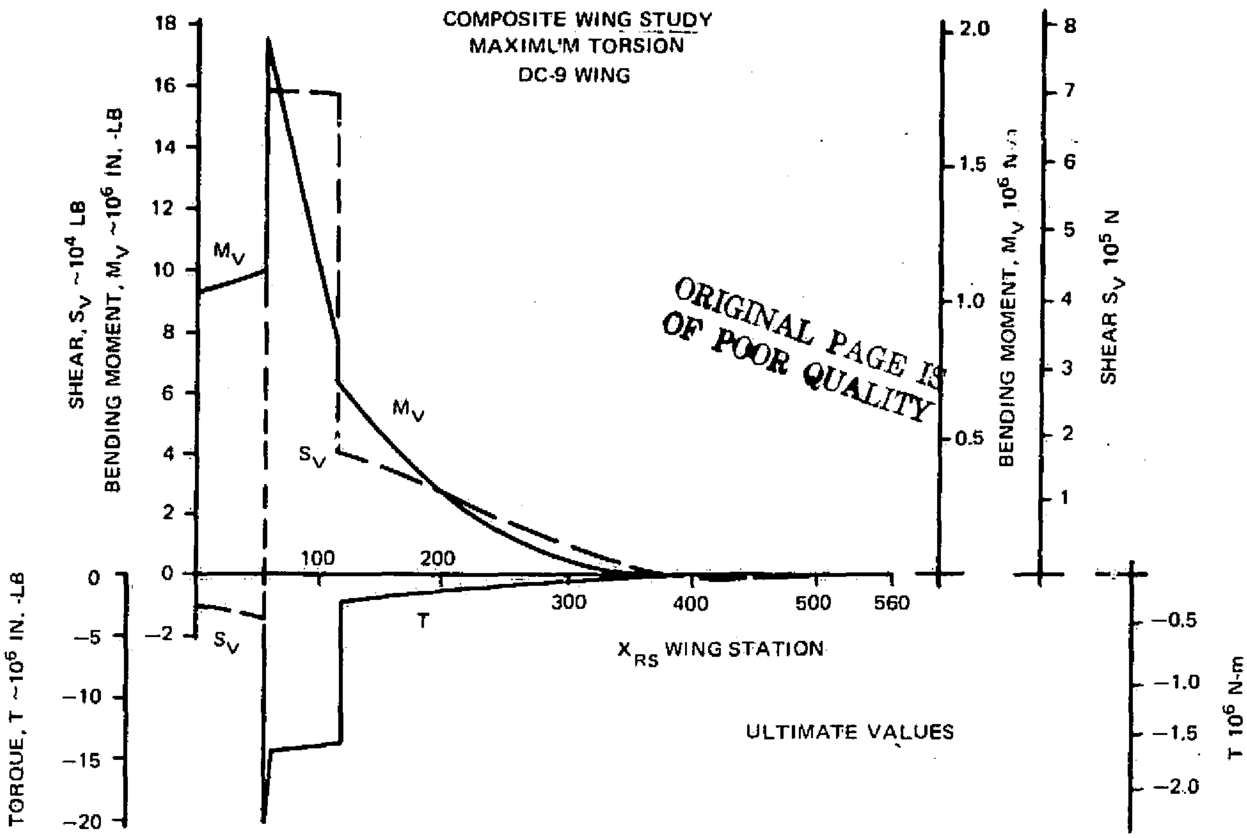


FIGURE 6-2. MAJOR WING BOX LOADING



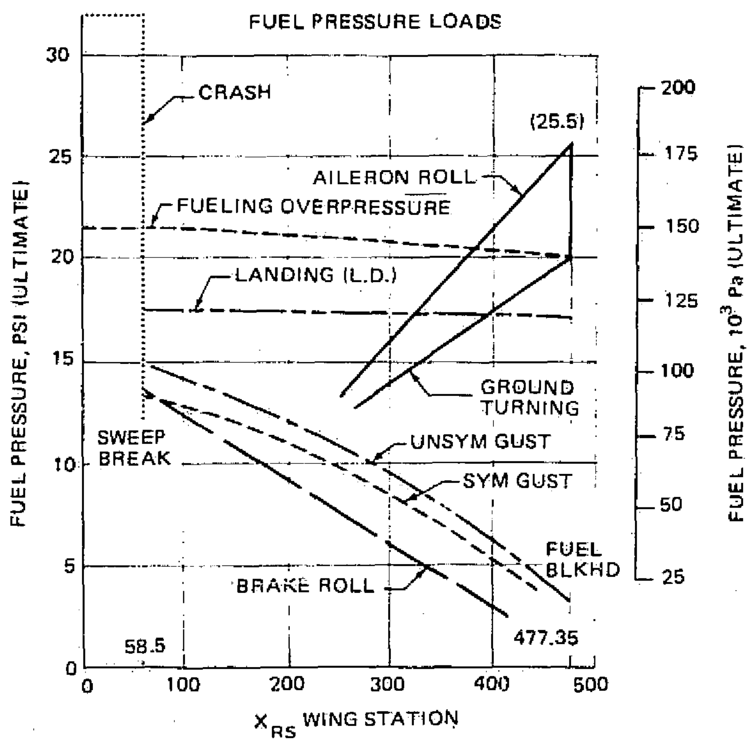


FIGURE 6-2. MAJOR WING BOX LOADING (CONT)

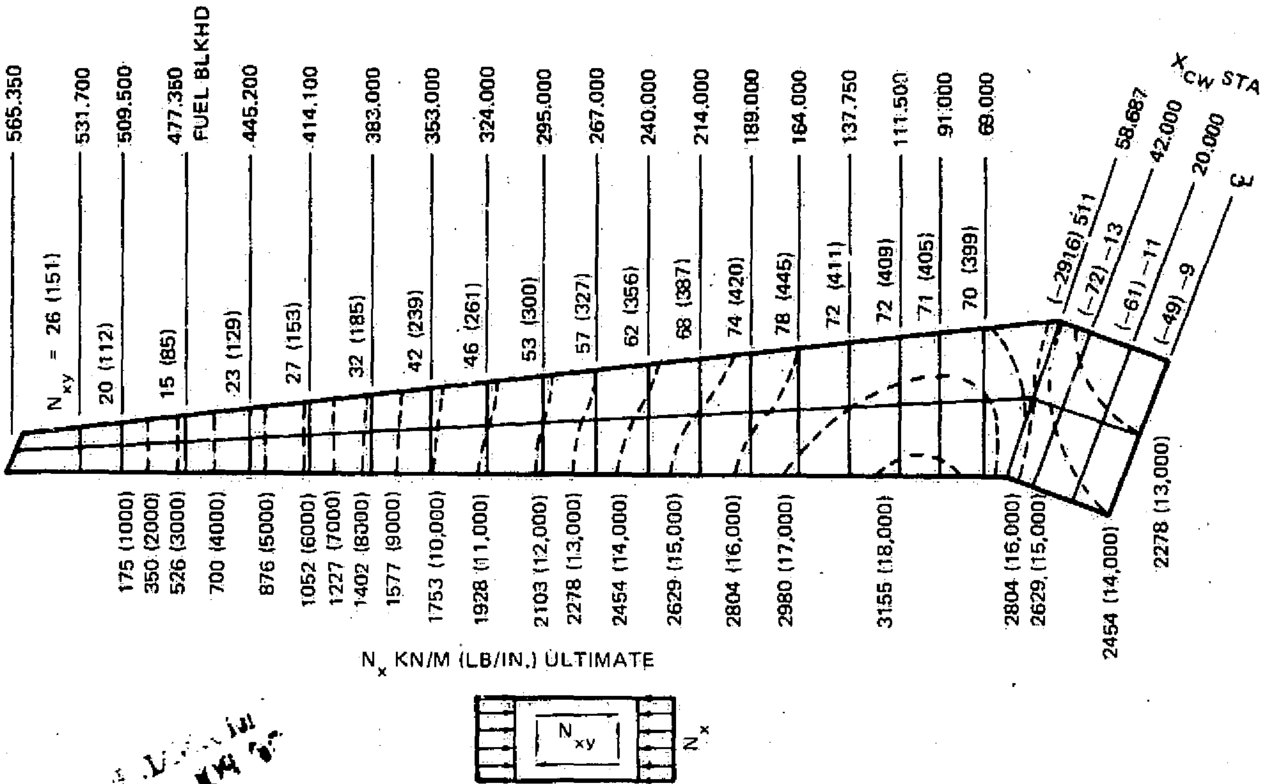
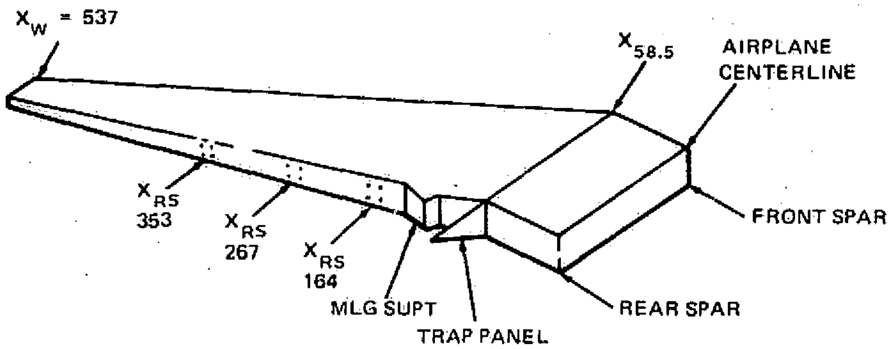
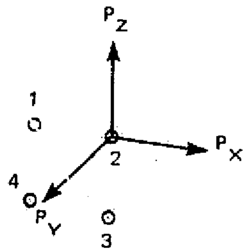


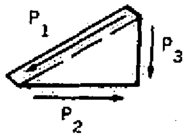
FIGURE 6-3. SKIN PANEL ULTIMATE LOADS



	FLAP HINGE ULTIMATE LOADS						
	STUD NUMBER	P_Y 10^3N	P_Y (10^3 LB)	P_Z 10^3N	P_Z (10^3 LB)	P_X 10^3N	P_X (10^3 LB)
X_{PS} 164	1	−221.0	(−49.6)	161.0	(36.3)	−60.9	(−13.7)
	2	−246.0	(−55.3)	140.0	(31.4)	−63.6	(−14.3)
	3	402.0	(90.3)	−133.0	(−29.8)	67.6	(15.2)
	4	129.0	(28.9)	−57.4	(−12.9)	67.2	(15.1)
X_{RS} 267	1	−195.0	(−43.9)	77.0	(17.3)	−39.1	(−8.8)
	2	−174.0	(−39.1)	64.1	(14.4)	−41.4	(−9.3)
	3	328.0	(73.8)	−9.8	(−2.2)	50.3	(11.3)
	4	108.0	(24.2)	−64.9	(−14.6)	29.8	(6.7)
X_{PS} 353	1	−36.9	(−8.3)	11.1	(2.5)	−15.1	(−3.4)
	2	−74.7	(−16.8)	3.6	(0.8)	−15.1	(−3.4)
	3	87.2	(19.6)	11.1	(2.5)	17.4	(3.9)
	4	49.4	(11.1)	3.6	(0.8)	17.4	(3.9)



	TRAP PANEL ULTIMATE LOAD					
COND	P_1 10^3N	P_1 (10^3 LB)	P_2 10^3N	P_2 (10^3 LB)	P_3 10^3N	P_3 (10^3 LB)
SU	209.9	(47.2)	697.9	(156.9)	447.0	(100.5)
HVL	205.1	(46.1)	762.4	(171.4)	357.6	(80.4)
LD	475.5	(106.9)	686.8	(154.4)	370.1	(83.2)
BP	287.8	(64.7)	625.8	(140.7)	343.4	(77.2)
HVL (1.16)	238.0	(53.5)	884.3	(198.8	415.0	(93.3)
LD (1.16)	429.3	(124.0)	796.6	(179.1)	429.2	(96.5)



MLG ATTACH FITTING ULT LOADS

AFT TRUNNION				FORWARD TRUNNION					
V_B 10^3N	V_B (10^3 LB)	S_B 10^3N	S_B (10^3 LB)	V_C 10^3N	V_C (10^3 LB)	D_C 10^3N	D_C (10^3 LB)	S_C 10^3N	S_C (10^3 LB)
2045.0	(459.8)	−116.1	(−26.1)	−1508.0	(−339.0)	443.9	(99.8)	12.0	(2.7)
−886.0	(−199.2)	−32.5	(−7.3)	987.5	(220.0)	−264.2	(−59.4)	101.4	(22.8)
1931.0	(434.0)	96.5	(21.7)	−1592.0	(−357.9)	443.9	(99.8)	−136.1	(−30.6)

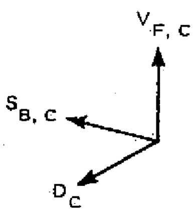
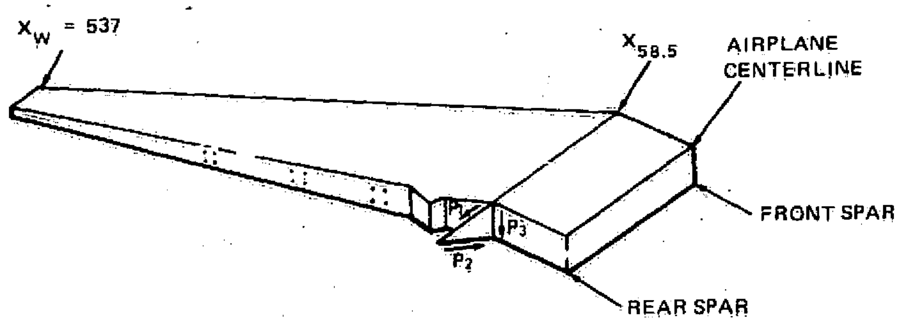
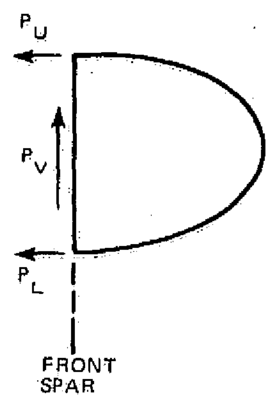


FIGURE 6-4. MAJOR SUPPORT LOADS



LEADING EDGE SLAT RIB LOADS

TRACK SUPPORT RIB NO.	SLAT NO.	STATION X_{IS} X_{OS}	P_U 10^3N	P_U (10^3 LB)	P_L 10^3N	P_L (10^3 LB)	P_V 10^3N	P_V (10^3 LB)
1. IDLER	1	82	−72.9	(−16.4)	90.7	(20.4)	40.0	(9.0)
2. DRIVER	1	135	37.4	(8.4)	−45.7	(−10.5)	12.9	(2.9)
2.	1	138	−32.5	(−7.3)	55.6	(12.5)	21.8	(4.9)
3. IDLER	1	186	−13.8	(−3.1)	20.9	(4.7)	14.7	(3.3)
4. IDLER	2	197	−10.7	(−2.4)	14.2	(3.2)	6.2	(1.4)
5. DRIVER	2	242	−37.8	(−8.5)	45.8	(10.3)	23.1	(5.2)
5.	2	233	−38.3	(−8.6)	46.3	(10.4)	43.6	(9.8)
6. COMMON IDLER	2. 3	285	−17.3	(−3.9)	22.2	(5.0)	9.3	(2.1)
7. DRIVER	3	335	−57.8	(−13.0)	79.6	(17.9)	35.1	(7.9)
8. COMMON IDLER	3 4	383	−26.7	(−6.0)	37.4	(8.4)	16.9	(3.8)
9. DRIVER	4	432	−62.7	(−14.1)	80.5	(18.1)	29.8	(6.7)
10. COMMON IDLER	4 5	485	−27.6	(−6.2)	37.8	(8.5)	16.5	(3.7)
11. DRIVER	5	534	−48.5	(−10.9)	64.1	(14.4)	21.4	(4.8)
12. IDLER	5	534	−22.7	(−5.1)	30.2	(6.8)	9.8	(2.2)



AILERON SUPPORT BOLT LOADS AT REAR SPAR

ULTIMATE VALUES

	BOLT	P 10^3N	P (10^3 LB)	V 10^3N	V (10^3 LB)	H 10^3N	H (10^3 LB)	CONDITION
HINGE NO. 1 X_{RS} = 383	UPR	15.0	(3.38)	−6.54	(−1.47)	3.78	(0.85)	ABRUPT ROLL
	UPR	−21.1	(−4.74)	6.32	(1.42)	−5.43	(−1.22)	STEADY ROLL
	LWR	19.8	(4.46)	1.56	(0.35)	4.98	(1.12)	STEADY ROLL
	LWR	−13.3	(−2.98)	−0.67	(−0.15)	−2.71	(−0.61)	STEADY ROLL
HINGE NO. 1 X_{RS} = 391	UPR	−10.5	(−2.36)	0	(0)	2.09	(−0.47)	ABRUPT ROLL
	LWR	8.14	(1.80)	0.9	(0.2)	−3.91	(−0.88)	ABRUPT ROLL
HINGE NO. 2 X_{RS} = 414	UPR	70	(15.8)	−21.3	(−4.79)	–	–	ABRUPT ROLL
	UPR	−71.1	(−16.0)	21.5	(4.84)	–	–	ABRUPT ROLL
	LWR	76.1	(17.1)	5.52	(1.24)	–	–	ABRUPT ROLL
	LWR	−68.5	(−15.4)	−4.98	(−1.12)	–	–	ABRUPT ROLL
HINGE NO. 3 X_{RS} = 477	UPR	−69.8	(−15.7)	−17.8	(−4.01)	–	–	ABRUPT ROLL
	UPR	−63.2	(−14.2)	16.1	(3.62)	–	–	ABRUPT ROLL
	LWR	70.7	(15.9)	4.49	(1.01)	–	–	ABRUPT ROLL
	LWR	−66.7	(−15.0)	−4.23	(−0.95)	–	–	ABRUPT ROLL

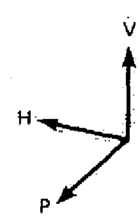


FIGURE 6-4. (CONTINUED)

and T-stiffened panels, shown in Figure 6-5, were evaluated on the basis of strength, stiffness, and weight.

The structural efficiency of these concepts unconstrained by stiffness criteria (i.e., for strength only) was evaluated as presented in Figure 6-6. Here, curves showing weight variation with compressive load intensity indicate significant differences in structural efficiency, with blade-stiffened panels being the least efficient. However, further investigation showed that the panel area required to satisfy bending and torsional stiffness criteria was the same in all cases, and greater than the area required by strength. This resulted in equal weights for all concepts. Blade-stiffening was selected as the least complex for fabrication and thus the one which would minimize costs.

Three candidate configurations shown in Figure 6-7 were evaluated for the spar and rib conceptual design. These configurations are of blade-stiffened, sine wave, and sandwich construction. The structural efficiency advantages of the sime wave and sandwich panels were offset by the structural provisions for subsystem installation and access: (1) subsystems — control, fuel, hydraulic, and electrical, and (2) access — assembly, inspection, maintenance, and repair. The lower-cost blade-stiffened concept was selected since it would result in fewer changes in subsystem installation design and structural interface while maintaining the same access provisions as the metal wing at no appreciable weight increase.

STRUCTURAL DESCRIPTION

The DC-9 wing consists of a main structural box which forms the fuel tank and supports the fixed leading edge and slats off the front spar and the fixed trailing edge panels, aileron, spoilers, flap, and main landing gear off the rear spar, as shown in Figures 6-8 and 6-9. The structural box continuity is maintained across the fuselage. It is joined to the fuselage at the wing-fuselage intersection (X_{cw} = 58.500). The sweepbreak is also at this location, as noted in Figure 6-8.

The conceptual design of composite application to this wing is limited to the main structural box on the left-hand side. Attaching structure and control surfaces are assumed to be of existing aluminum construction. The

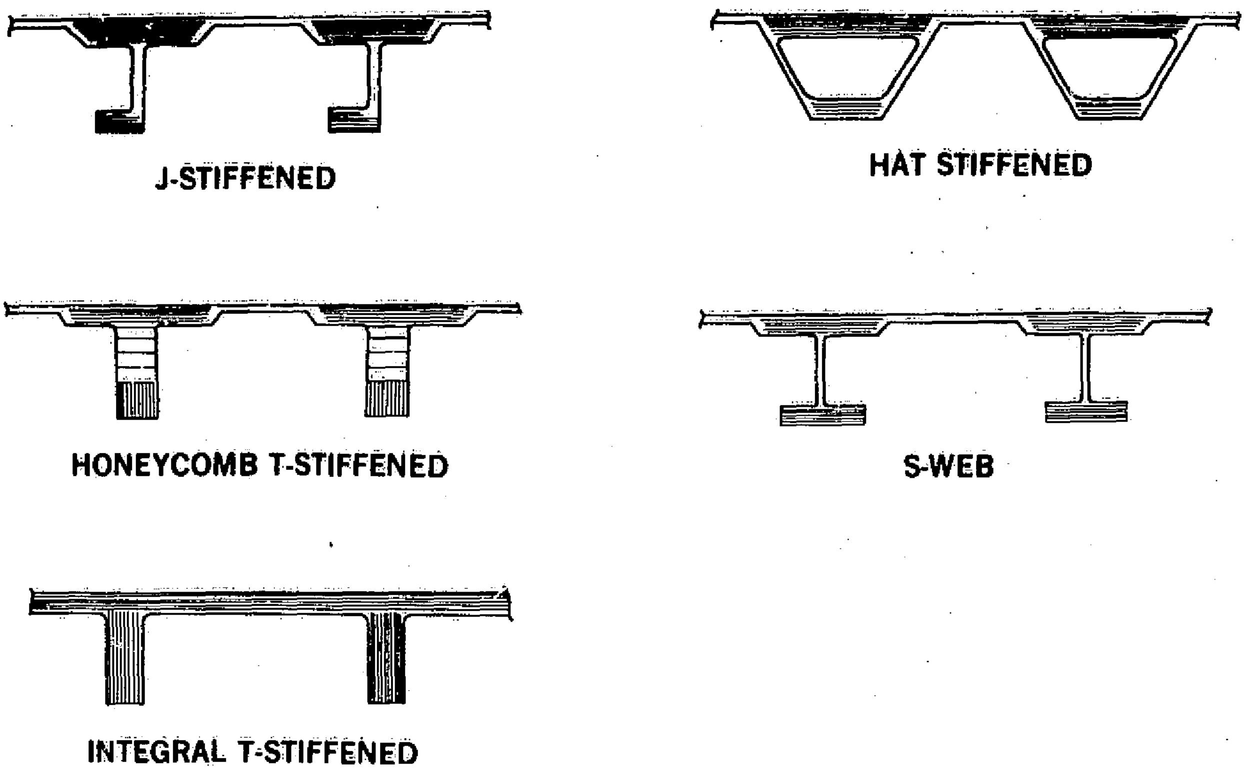


FIGURE 6-5. COVER PANEL CONFIGURATIONS

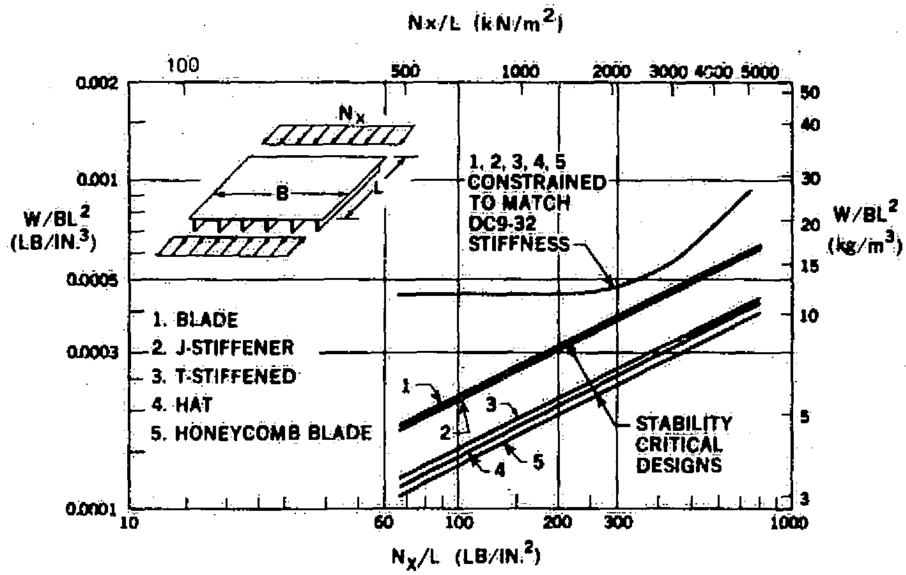


FIGURE 6-6. STRUCTURAL EFFICIENCY COMPARISONS OF COVER PANEL DESIGNS

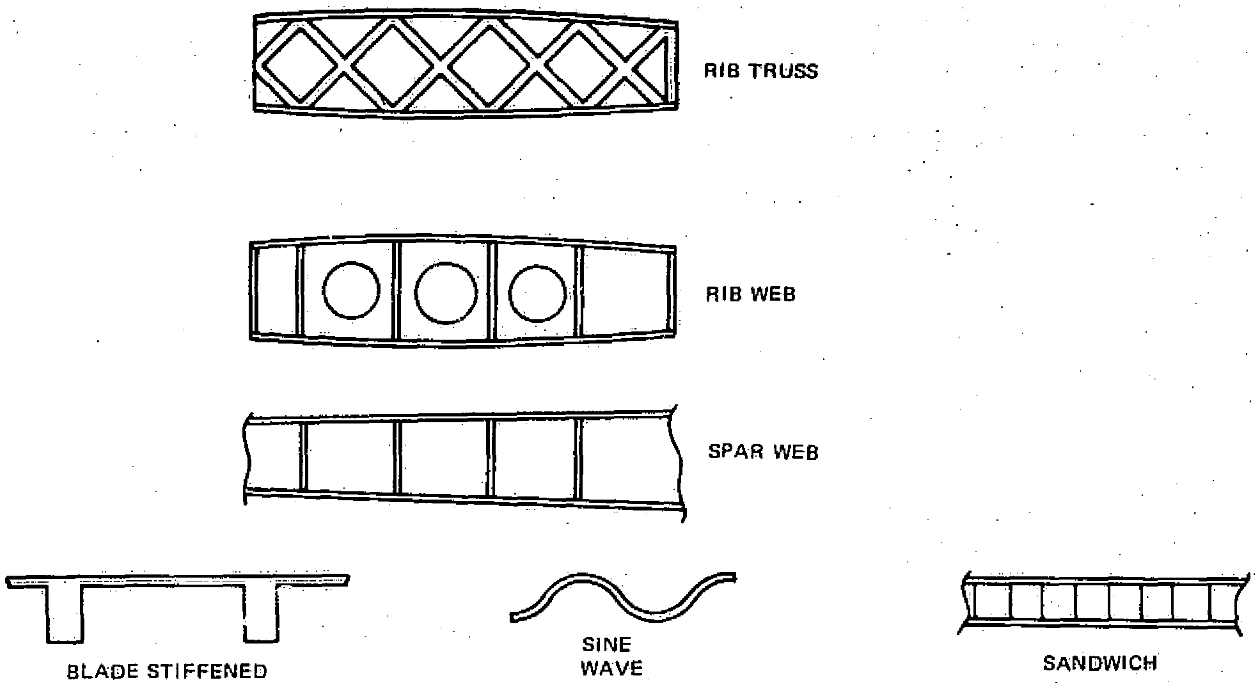


FIGURE 6-7. CONCEPTUAL DESIGN SPAR AND RIB CONFIGURATIONS

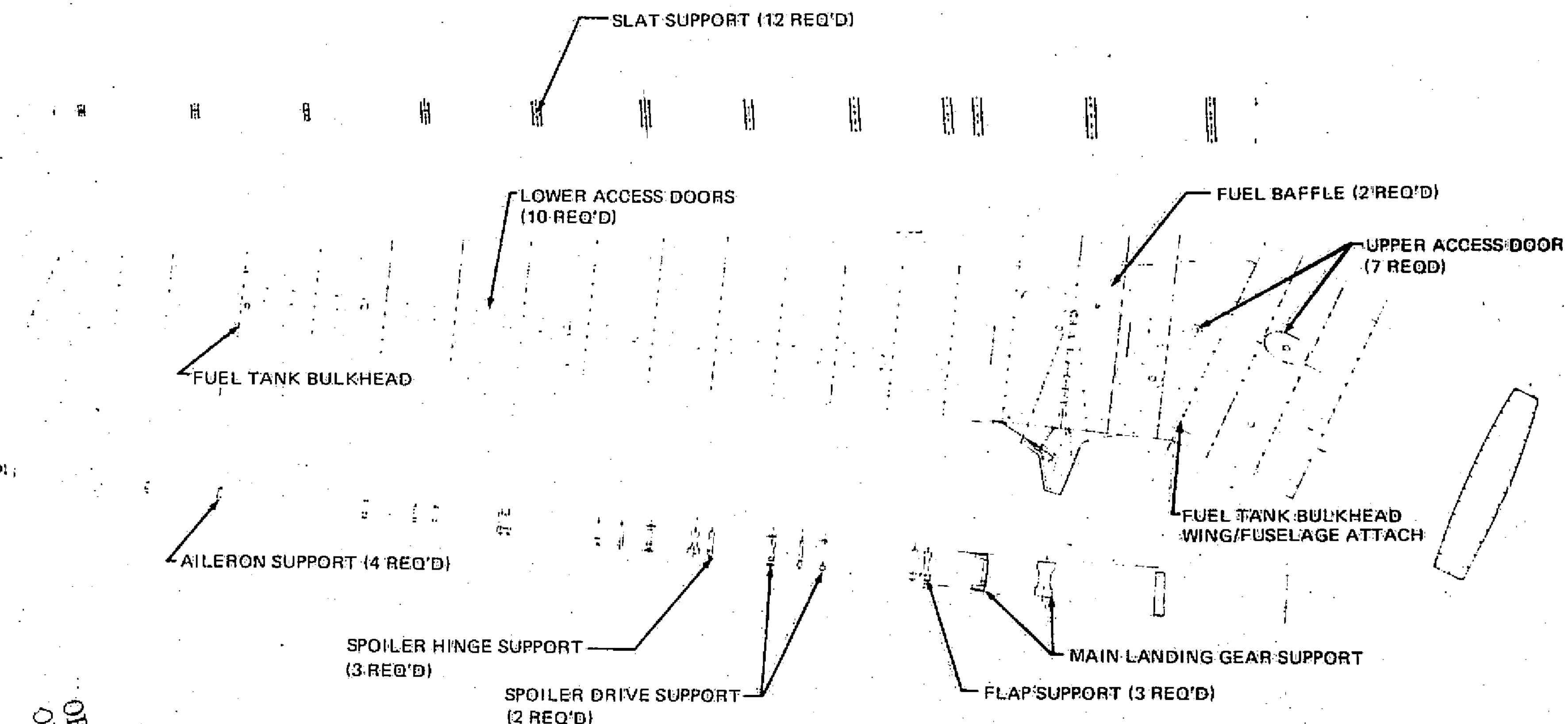


FIGURE 6-8. DC-9 COMPOSITE WING GENERAL ARRANGEMENT

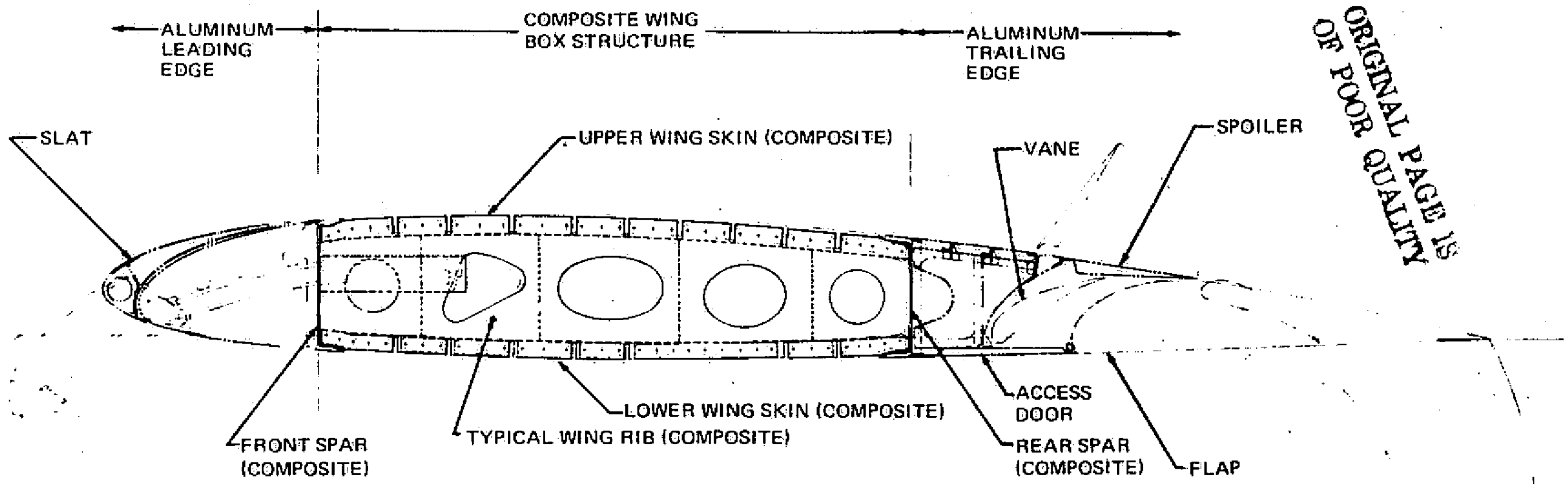


FIGURE 6-9. TYPICAL WING CROSS-SECTION

composite left-hand box is joined to an aluminum right-hand box at the centerline of the airplane and to the fuselage at the sidewall.

The composite box is approximately 15.25 meters (50 feet) long, 3.5 meters (12 feet) wide, and 0.5 meter (2 feet) deep. It is a two-spar, multirib arrangement with each component in the same location as its metal counterpart. Each composite member performs the same function and resists the same loads as the aluminum member it replaces.

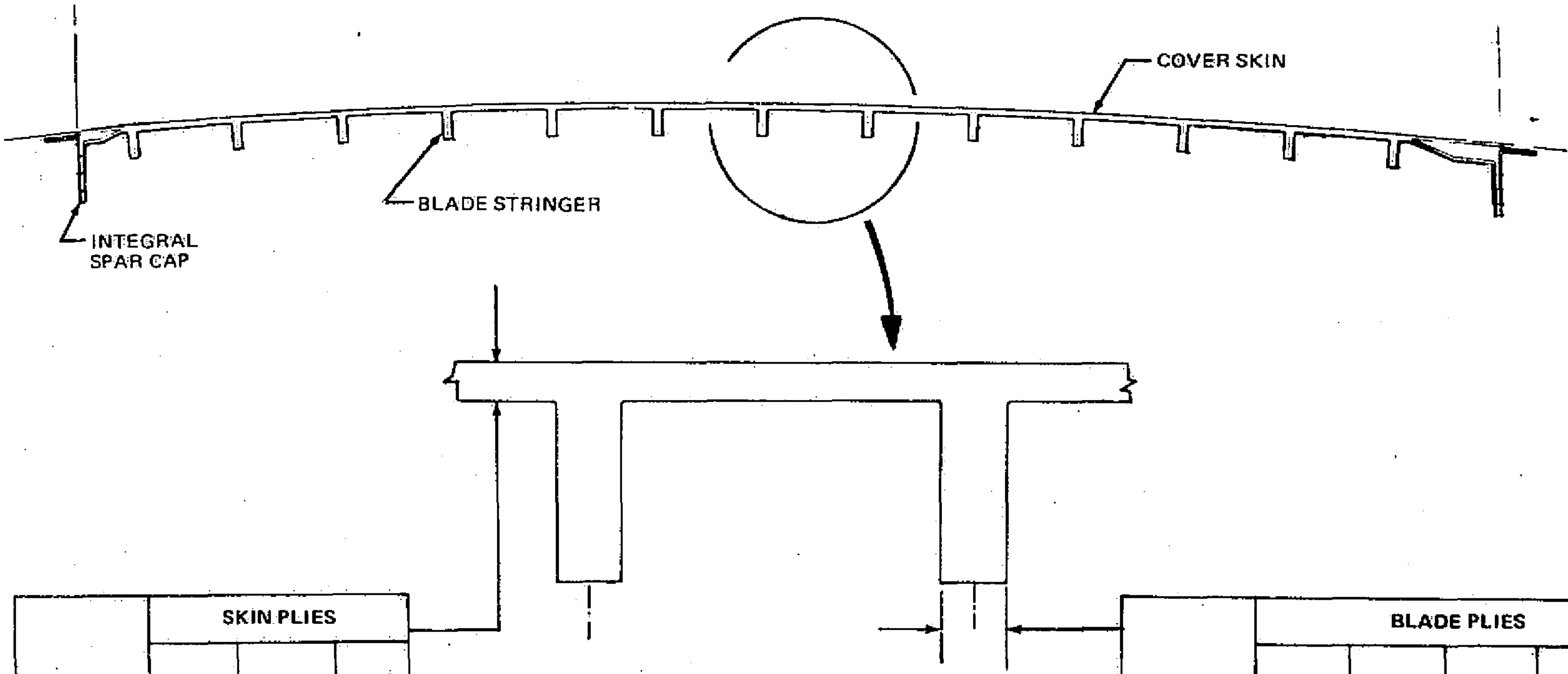
Wing Skin Panel

The shear-resistant wing cover panels are composed of graphite/epoxy laminated skins with integrally stiffened blade stringers, as shown in Figure 6-10. Composite spar caps are interlaminated and cocured with this wing panel. Figure 6-11 details a typical access cover installation that is provided in the skin to allow access to the wing box for assembly as well as fuel tank inspection and maintenance.

The blade stringers are in the same location on the lower wing panel as on the aluminum DC-9 Series 30 wing to facilitate installation of existing fuel pumps and their associated plumbing. The upper panel blade stringers were relocated to prevent skin panel buckling at design ultimate load. These blade stringers as well as the wing skins are tapered outboard to the wing tip bulkhead, as shown in Figure 6-10. Slotted holes are designed into the wing panel blade stringers to eliminate fuel entrapment. Cocured shear clips nested between the blade stringer attach the wing panel to the rib webs. These shear clips also allow passage of fuel through the rib boundary. Access through the wing skin is required for fuel venting, dipsticks (fuel quantity gages), and fuel probes. Their approximate number and location are given in Figure 6-8.

Wing Spar

Two major load-carrying members of the wing are the wing front and rear spars. Figure 6-12 details the spar configuration and its relation to the wing panel and indicates the laminate construction. The composite spar cap which is cocured to the wing skin is mechanically attached with titanium fasteners to a blade-stiffened 15-meter (50-foot) long composite spar web,



STATION	SKIN PLIES 0°	45°	90°
0.00			
	11	26	4
111.500			
	11	26	4
137.000			
	10	24	0
189.000			
	9	24	0
267.000			
	9	24	0
353.000			
	9	24	0
414.400			
	8	20	0
477.350			
	6	16	0
509.500			
	4	10	0
536.861			

STATION	BLADE PLIES 0°	45°	90°	HEIGHT
0.00				1.50
	64	18	0	
137.000				1.50
	60	14	0	
240.000				1.50
	54	12	0	
295.000				1.50
	44	12	0	
383.000				1.38
	38	12	0	
445.000				1.25
	28	12	0	
477.350				1.187
	24	12	0	
509.500				1.11
	24	12	0	
536.861				1.00

FIGURE 6-10. COMPOSITE WING SKIN PANELS

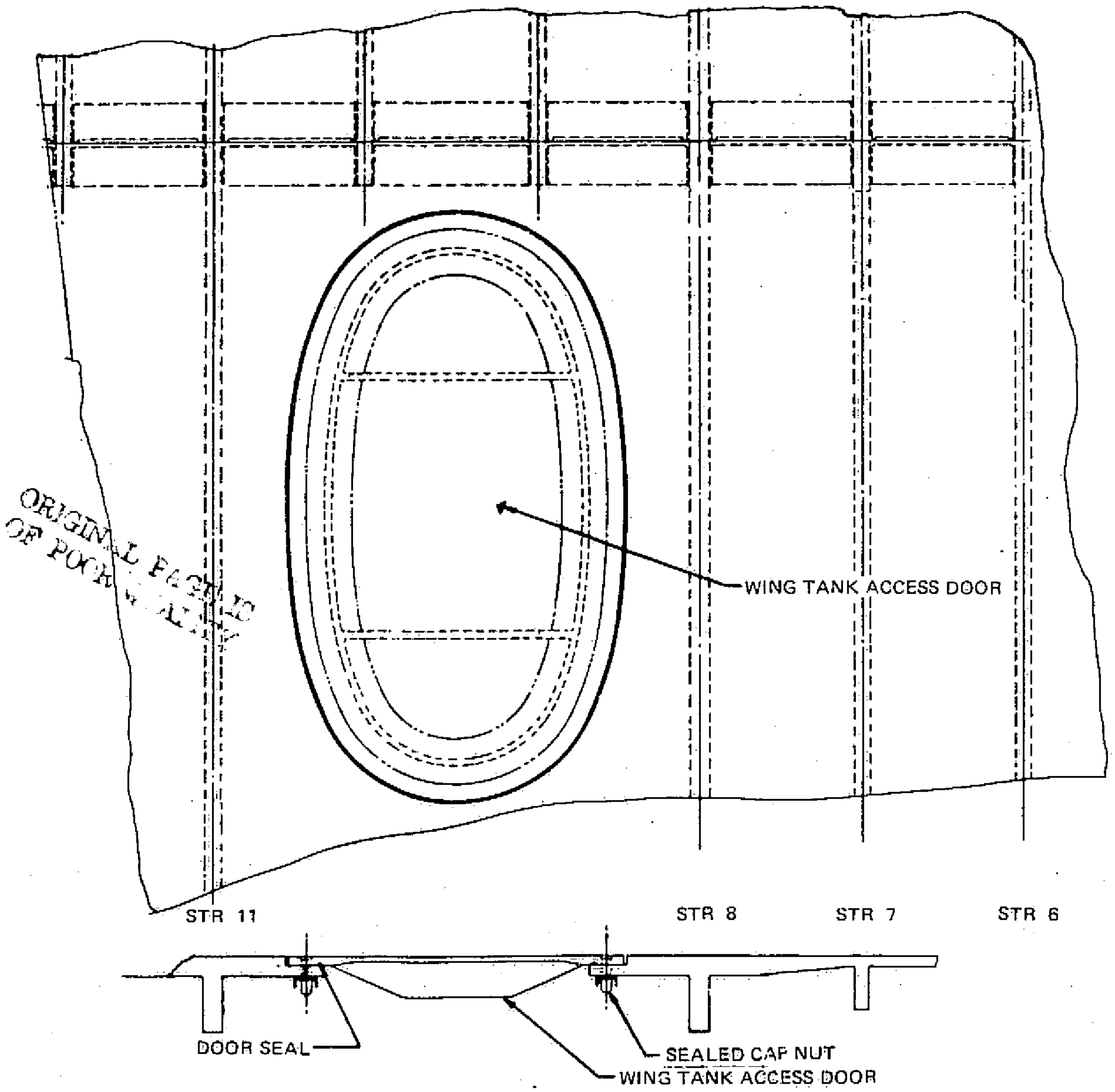


FIGURE 6-11. TYPICAL ACCESS COVER INSTALLATION

as shown in Figure 6-13. Highly loaded areas of the spar web such as the landing gear fitting and trapezoidal fitting are reinforced with additional laminates.

Typical Wing Rib

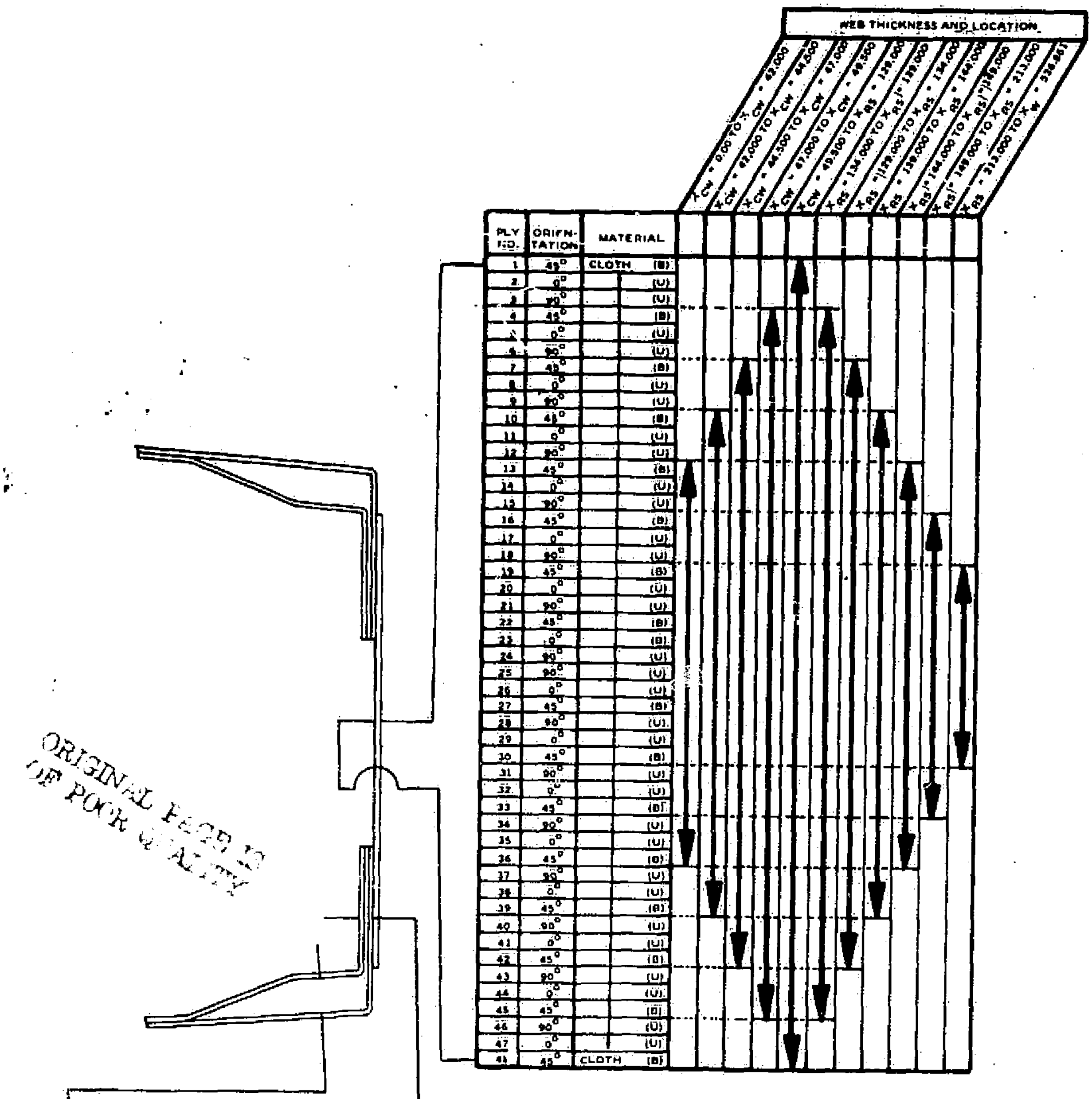
The composite wing rib as shown in Figure 6-14 is composed of a shear-resistant web with integrally cocured, vertically running stiffeners and an integrally formed spar attach tee. The web is attached mechanically to the wing skin by means of shear clips cocured to the skin as previously noted. Lightening holes in the web reduce weight and allow access to the wing assembly components and routing of the fuel piping system.

Baffled Fuel Bulkhead

This composite bulkhead, as shown in Figure 6-15, provides a barrier in the fuel tank to reduce the hydrostatic pressure at the end bulkheads and at the front and rear spars that could occur under certain flight maneuver conditions. The bulkhead is a sealed barrier with flapper doors located at the lower wing skin and in the middle of the bulkhead web. These flapper doors allow fuel to pass freely inboard to the fuel pumps and not outboard. Access doors in the web are provided for bulkhead assembly and inspection. This bulkhead is similar in design to the wing rib.

Main Landing Gear Support Structure

This structure consists of the main landing gear support fitting that is bolted to the wing rear spar, special upper and lower wing skin doublers, and landing gear rib, as shown in Figures 6-16 and 6-17. The upper and lower titanium tapered doublers are cocured into the wing skins. These doublers provide enough attachment bearing material to allow the use of the existing forged aluminum main landing gear fitting as well as provide a load path to redistribute these concentrated loads into the wing box skins. The in-tank integrally machined aluminum rib stabilizes the wing box similar to other wing ribs and also distributes the loads from the main gear into the wing box. Vertical gear loads are sheared directly into the wing rear spar. The main gear fitting is given torsional rigidity and side load stability by the auxiliary spar members.



PLY NO.	ORIEN-TATION	MATERIAL	
1	45°	CLOTH	(B)
2	0°		(U)
3	90°		(U)
4	45°		(B)
5	0°		(U)
6	90°		(U)
7	45°		(B)
8	0°		(U)
9	90°		(U)
10	45°		(B)
11	0°		(U)
12	90°		(U)
13	45°		(B)
14	0°		(U)
15	90°		(U)
16	45°		(B)
17	0°		(U)
18	90°		(U)
19	45°		(B)
20	0°		(U)
21	90°		(U)
22	45°		(B)
23	0°		(B)
24	90°		(U)
25	90°		(U)
26	0°		(U)
27	45°		(B)
28	90°		(U)
29	0°		(U)
30	45°		(B)
31	90°		(U)
32	0°		(U)
33	45°		(B)
34	90°		(U)
35	0°		(U)
36	45°		(B)
37	90°		(U)
38	0°		(U)
39	45°		(B)
40	90°		(U)
41	0°		(U)
42	45°		(B)
43	90°		(U)
44	0°		(U)
45	45°		(B)
46	90°		(U)
47	0°		(U)
48	45°	CLOTH	(B)

STATION	REAR SPAR CAP PLIES			MAT'L
	0°	45°	90°	
X_{CW} = 0.000	64U	16B	32U	CLOTH
X_{CW} = 58.688	80U	20B	40U	
X_{RS} = 111.500	48U	12B	24U	
X_{RS} = 300.000	20U	5B	10U	
X_{RS} = 414.000	12U	3B	6U	
X_W = 538.861	12U	3B	6U	CLOTH

U UNIDIRECTIONAL WEAVE

B BIDIRECTIONAL WEAVE

STATION	REAR SPAR CAP LEG PLIES			MAT'L
	0°	45°	90°	
X_{CW} = 0.000	24U	6B	12U	CLOTH
X_{CW} = 58.688	36U	9B	18U	
X_{RS} = 111.500	20U	5B	10U	
X_{RS} = 213.000	8U	2B	4U	CLOTH
X_{RS} = 370.000	0	0	0	

FIGURE 6-12. COMPOSITE SPAR CONSTRUCTION

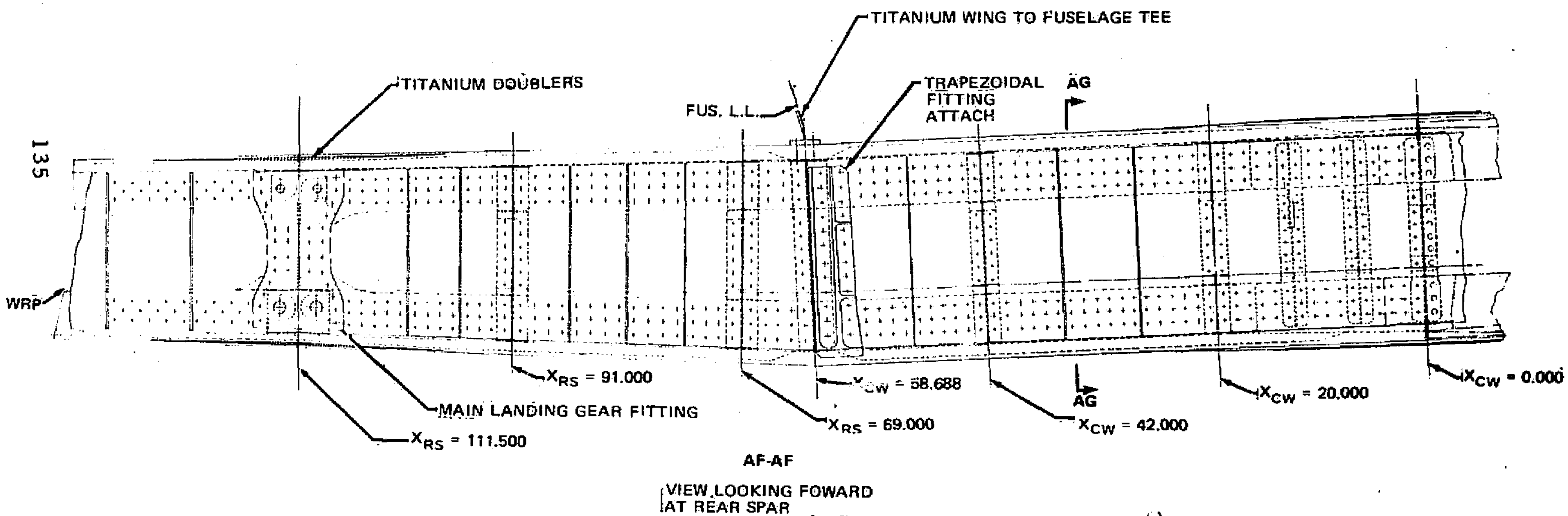


FIGURE 6-13. SPAR WEB INSTALLATION

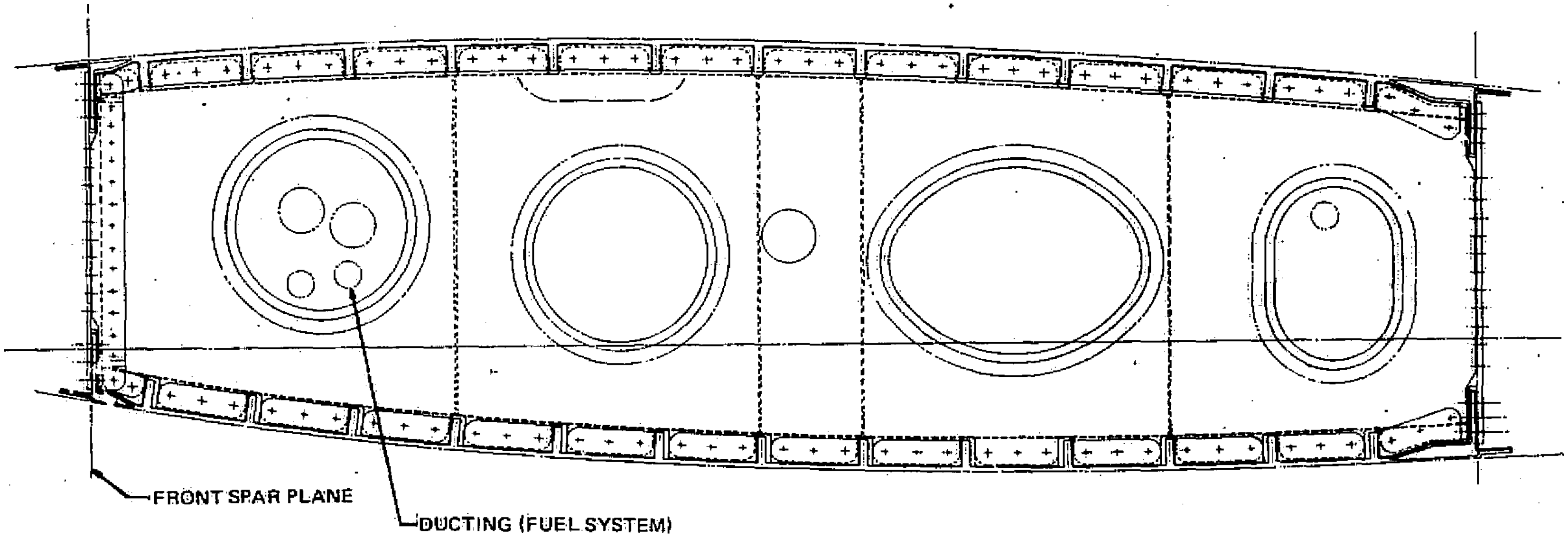


FIGURE 6-14. TYPICAL WING RIB

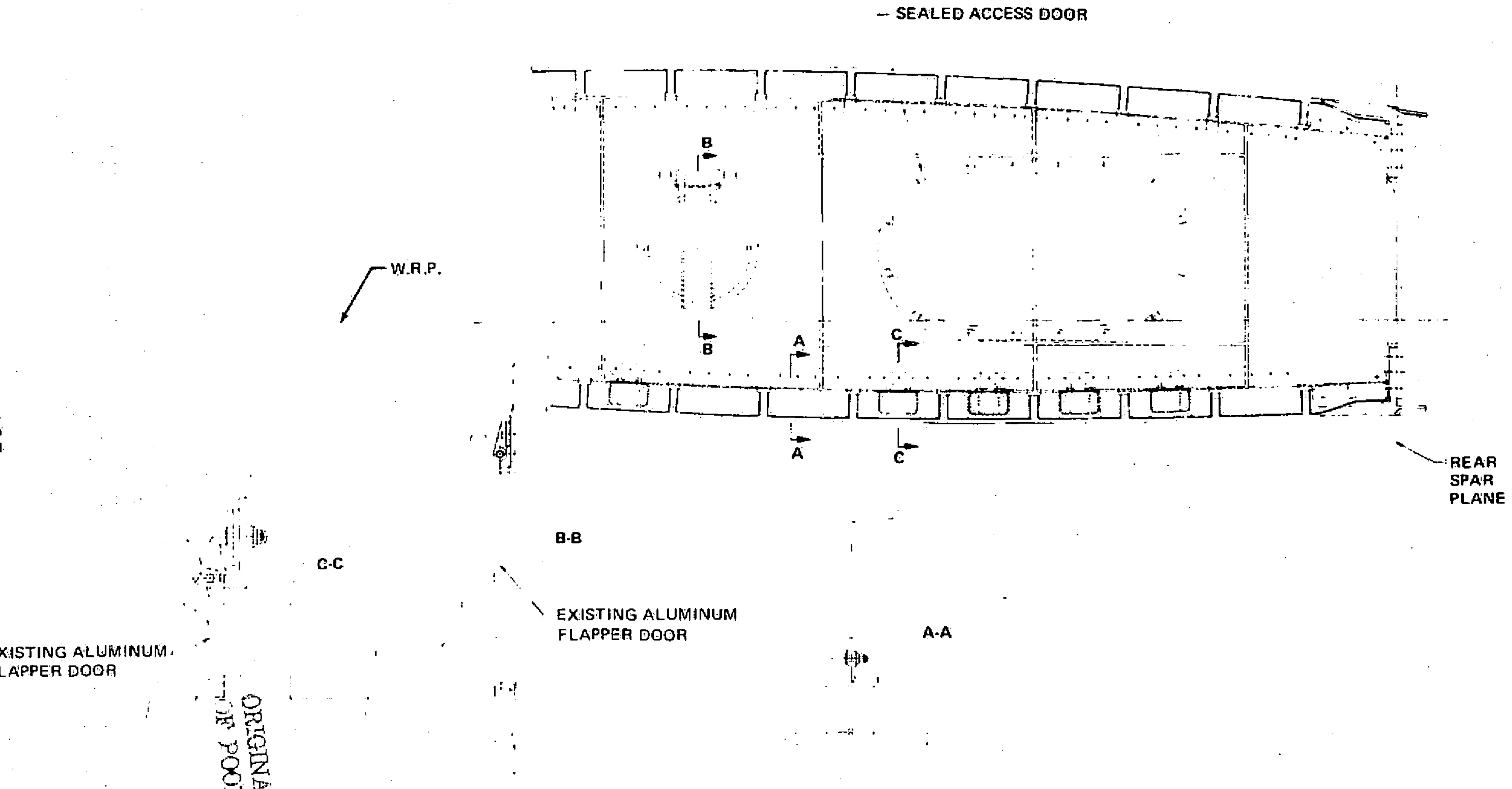


FIGURE 6-15. FUEL BAFFLE

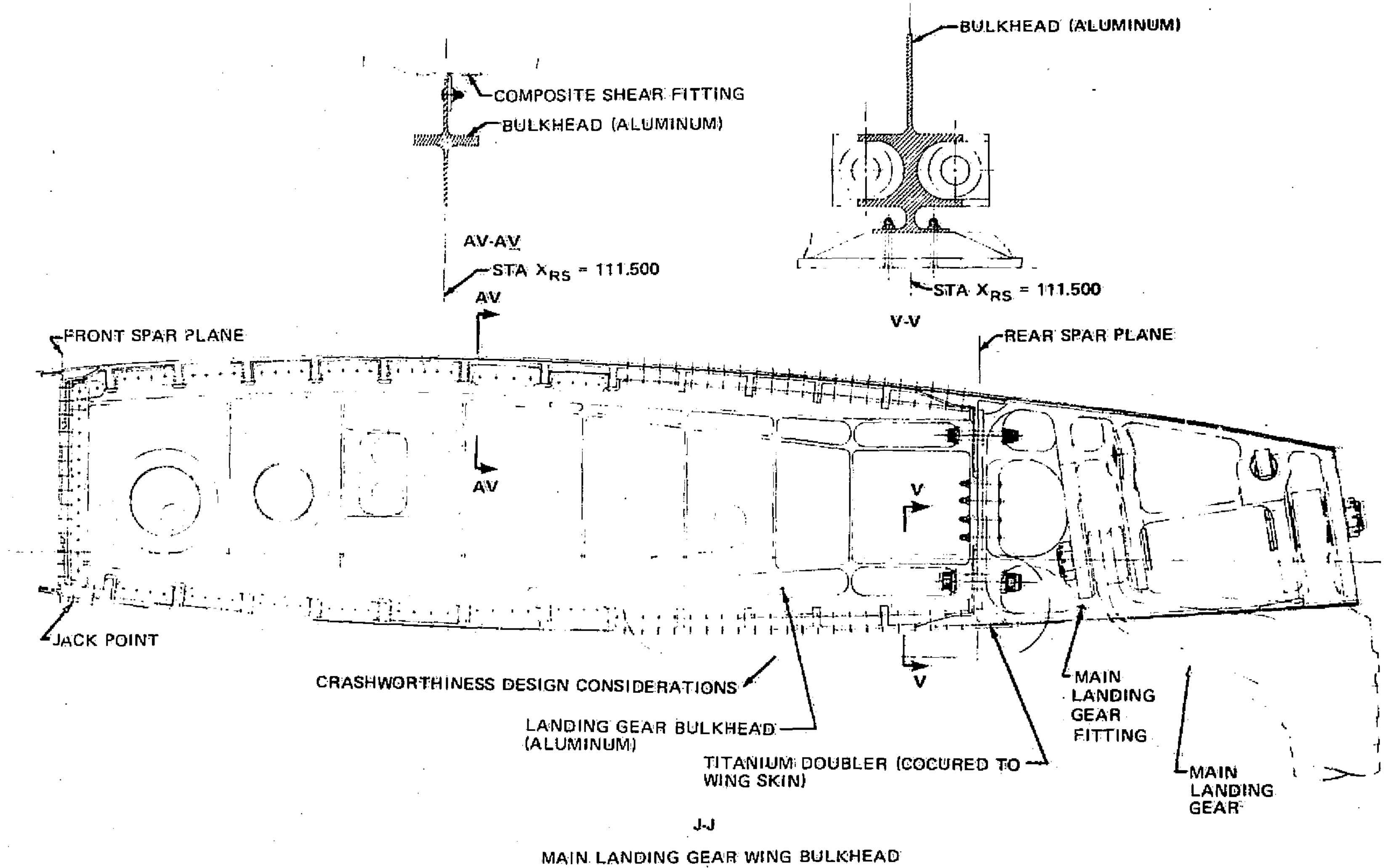


FIGURE 6-16. MAIN LANDING GEAR SUPPORT STRUCTURE

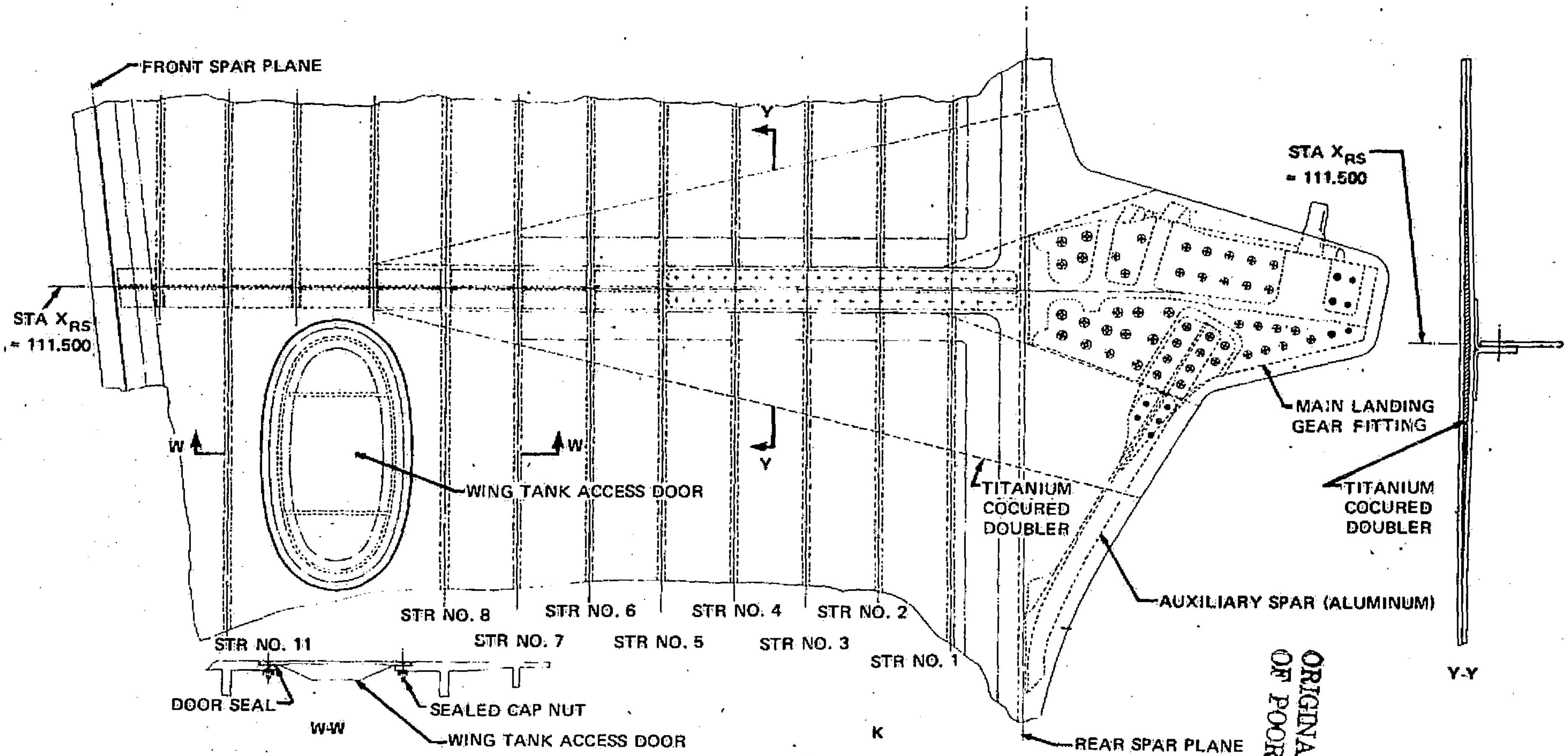


FIGURE 6-17. TOP VIEW OF MAIN LANDING GEAR SUPPORT STRUCTURE

Fuselage-Wing Intersection Bulkhead X_{CW} = 58.688

At the fuselage intersection, the wing skin blade stringers are blended into a thickened pad. This pad is interlaminated with a cocured composite blade to function as an attachment for the bulkhead web. The aluminum trapezoidal fitting is attached to this bulkhead immediately aft of the rear spar and a titanium tee ties the bulkhead and trapezoidal fitting to the fuselage skin panels, as shown in Figure 6-18. The primary function of the X_{CW} = 58.688 bulkhead is to transfer wing loads into the fuselage shell, stabilize the wing box structure, and provide a closing fuel tank barrier. The titanium tee located on top of the wing is designed to transfer wing horizontal shear into the fuselage, to maintain the cabin pressure boundary at the upper wing-fuselage intersection, and to provide structural continuity across the wing-fuselage cutout. Vertical loads are transferred from the rear spar into a specially machined fuselage frame by way of the trapezoidal fitting. This fitting transfers the landing gear retract link vertical loads into this same fuselage frame.

Aileron Hinge Support and Fuel Tank Bulkhead

This bulkhead, as shown in Figure 6-19, transfers the aileron loads introduced at the rear spar-mounted supporting fitting to the wing box structure. Titanium fittings are cocured into this bulkhead to transmit the load from the aileron. The bulkhead is also the outboard closing end for the wing fuel tank. To facilitate tank sealing, the wing blade stringers are faired into the wing skin under the bulkhead tee.

Flap Hinge Support Rib

The flap support rib is similar in design to the aileron bulkhead except it is not a fuel tank barrier. The wing blade stiffeners pass through the bulkhead plane and are connected to the bulkhead in the same manner as the intermediate ribs. Design details of the flap support rib are shown in Figure 6-20.

Center Wing Joint

This joint mechanically attaches the left-hand side composite wing to the right-hand side conventional aluminum wing through the use of metal splice plates and spar splice angles. Figure 6-21 shows a section cut at the airplane centerline. Figure 6-22 details how stringers are blended into lands or pads to minimize stress concentrations and are indirectly joined by conventional

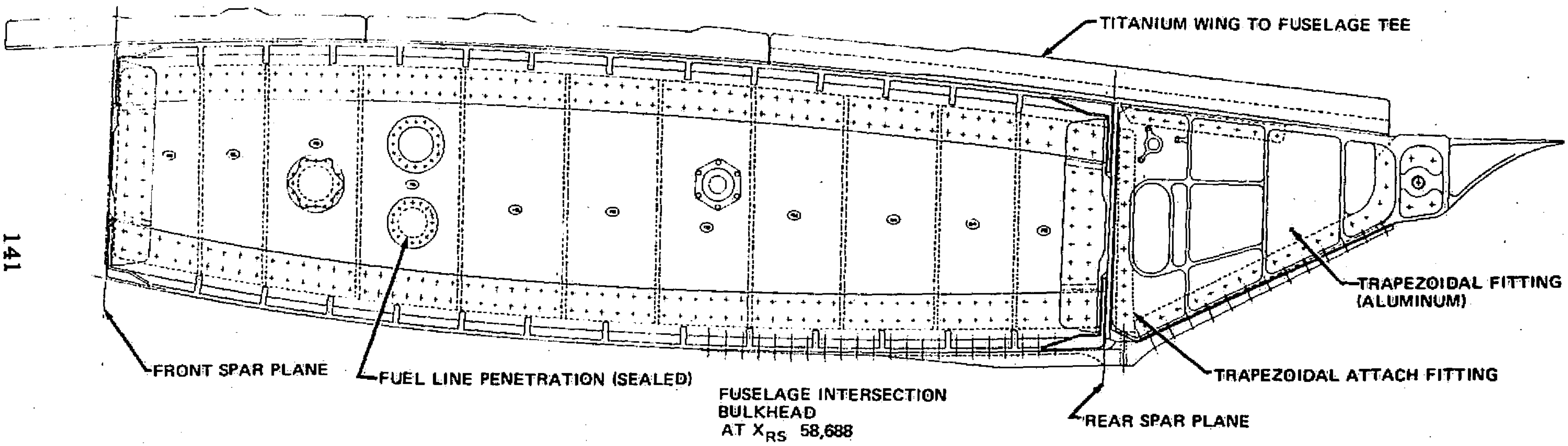


FIGURE 6-18. FUSELAGE/WING ATTACHMENT BULKHEAD

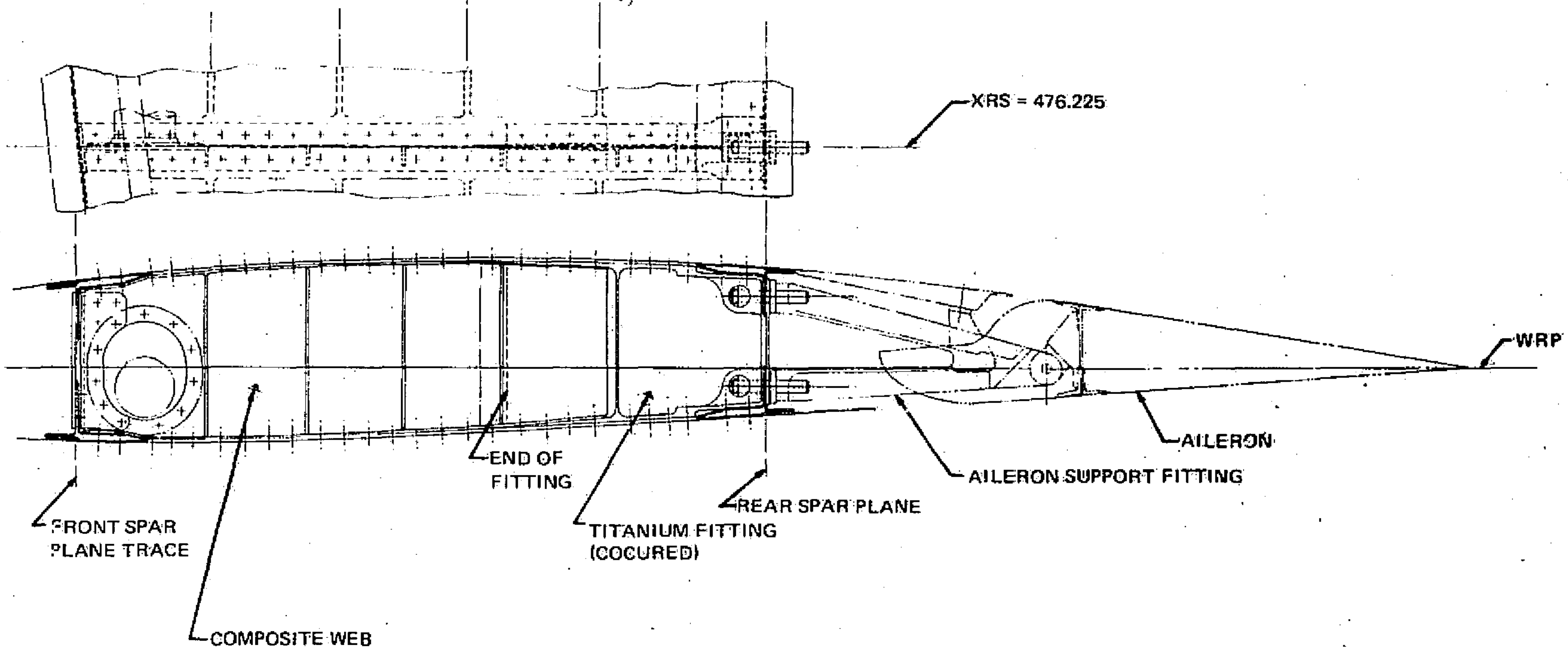


FIGURE 6-19. AILERON SUPPORT AND FUEL TANK BULKHEAD

butt splicing. The spar caps are tapered and joined with tapered splice angles and plates, as shown in Figures 6-23 and 6-24.

Wing Leading Edge

The fixed leading edge uses aluminum structure of the DC-9 Series 30 which is modified to adapt to the composite front spar, as shown in Figure 6-25. This structure consists of skin stiffened by chordwise ribs and formers that are permanently attached to the wing. Double ribs support the slat-track assembly as shown in Figure 6-9. Sealed cans supported off the front spar web encase the slat track to allow penetration into the fuel tank cavity.

Wing Trailing Edge

The aluminum trailing edge structure is located between the rear spar and control surfaces (spoiler, flaps, and ailerons), as shown in Figure 6-25.

WEIGHT ESTIMATE

A weight estimate of the conceptual design and a comparison with the existing metal wing box are presented in Table 6-1. The weight includes only those items that are functionally chargeable to wing structure. Not included are many small clips and fasteners on the spars and bulkheads which are used to assist the wing's secondary function as a fuel tank. Significant items that are not included in the weight estimate are the wing bulkheads at station X_{rs} = +111.5 which are charged to landing gear support structure. In all cases, the composite box weight includes exactly the same items that are in the metal design's weight.

A study was conducted to determine the effect of appropriate design constraints on the weight of the DC-9 composite wing box. In addition to the stiffness-limited conceptual design, weight estimates were made for two strain-limited configurations, both singly and in combination with stiffness limitations. Strain limitations may be required by damage tolerance considerations. These are compared with conventional strength critical or unconstrained designs. The criteria used in this study are as follows:

Stiffness — Maintain the same bending (EI) and torsional (JG) stiffness in the composite wing box as in the aluminum configuration.

Strain (0.004) — 0.004 in./in. maximum permissible strain in any component.

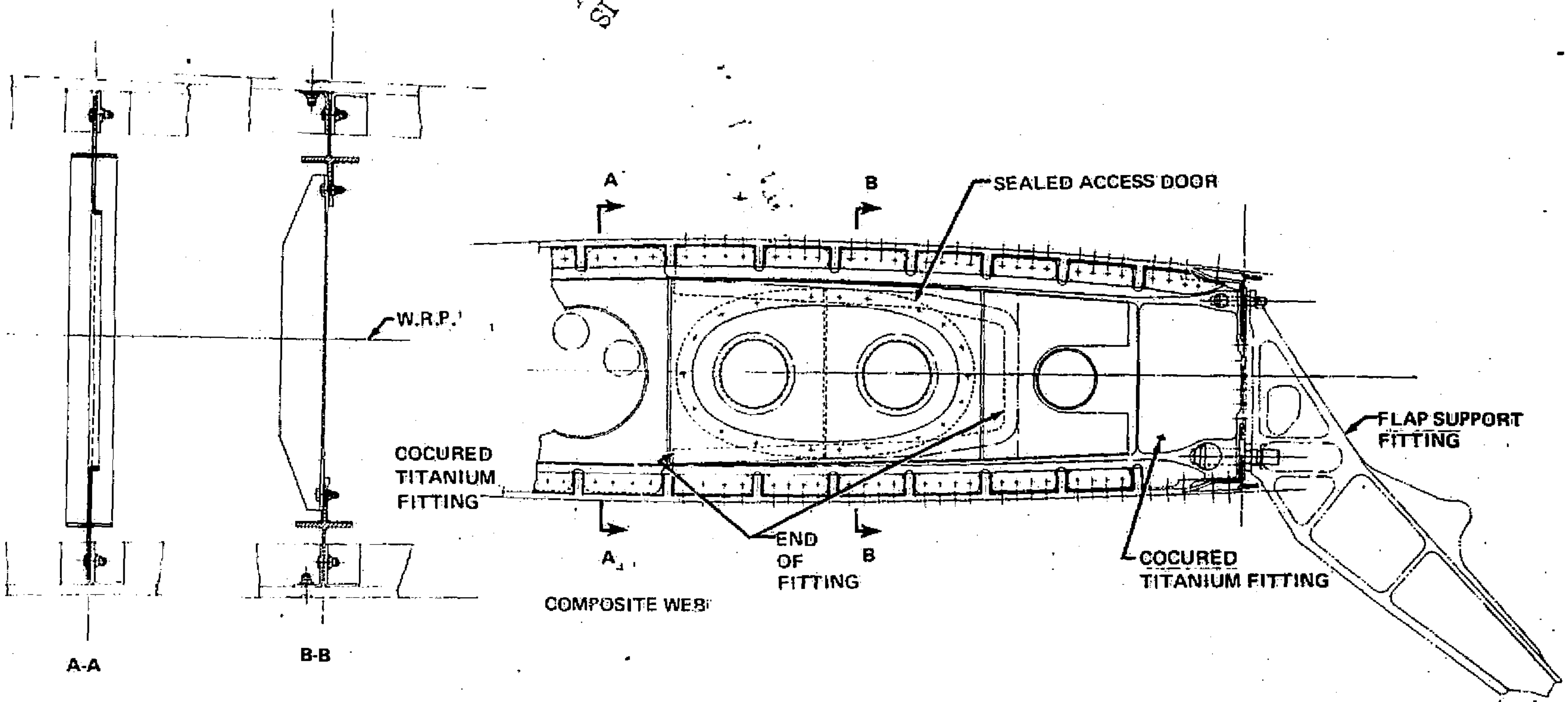


FIGURE 6-20. FLAP SUPPORT RIB

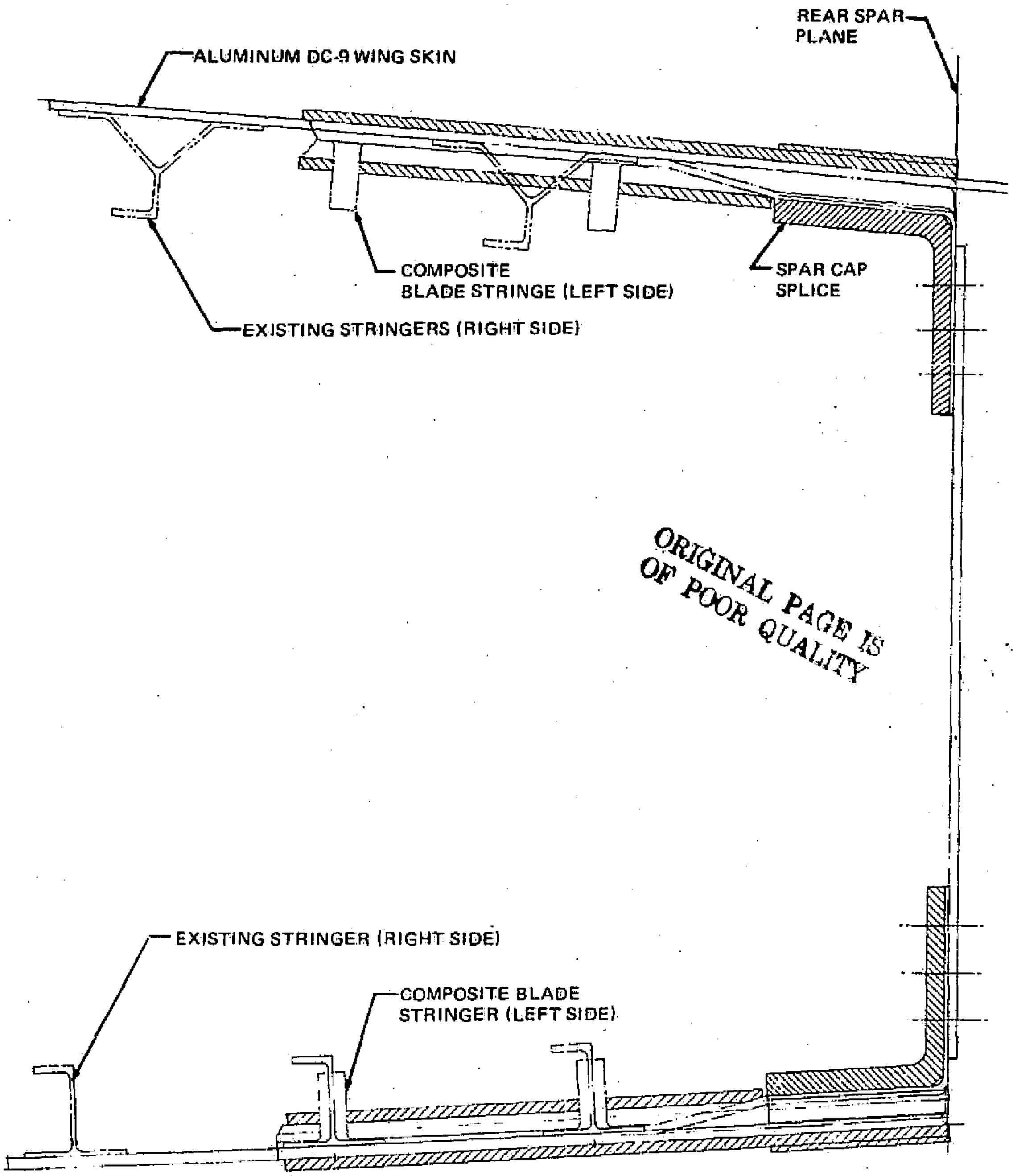


FIGURE 6-21. SECTION-CUT CENTERLINE WING SPLICE

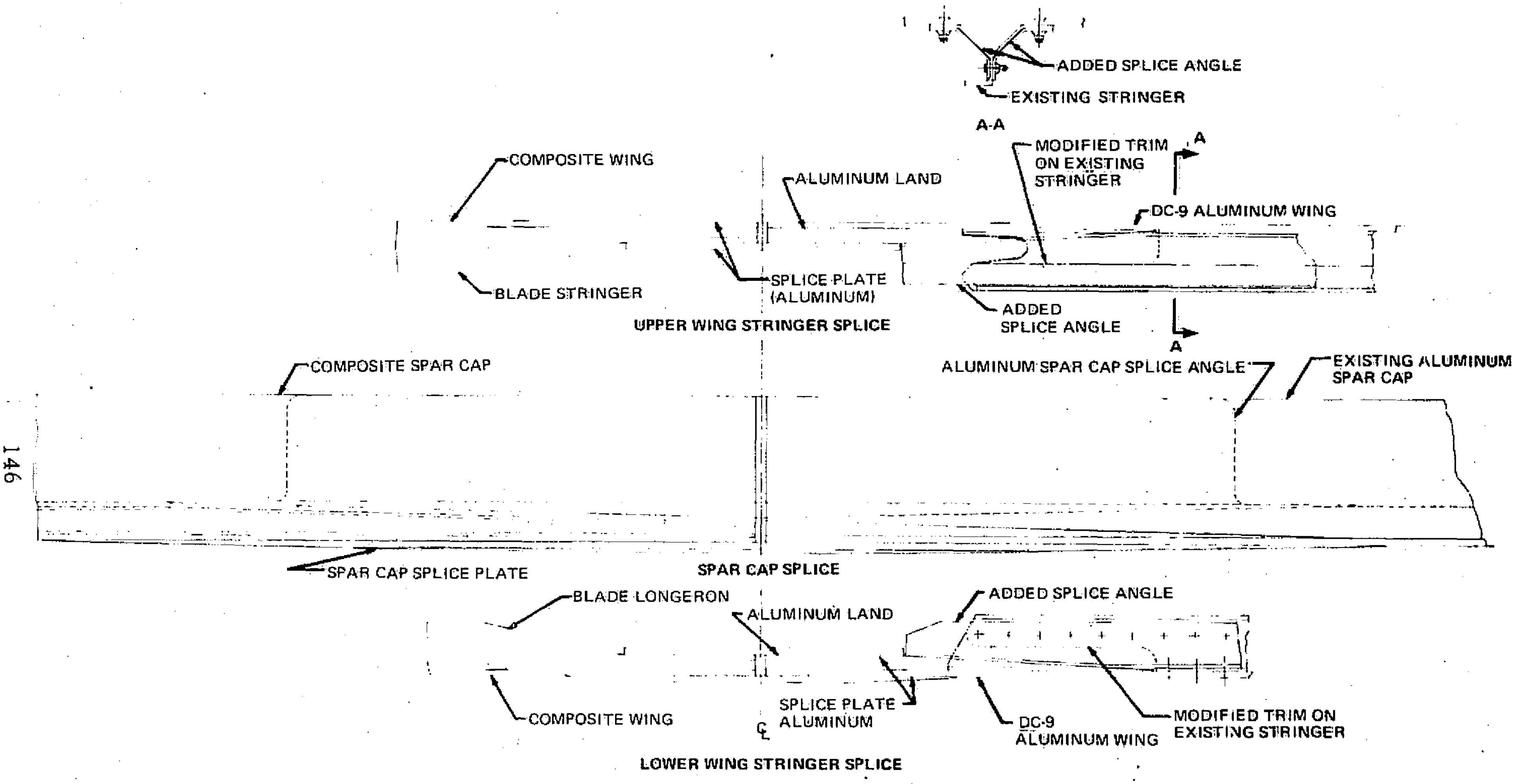
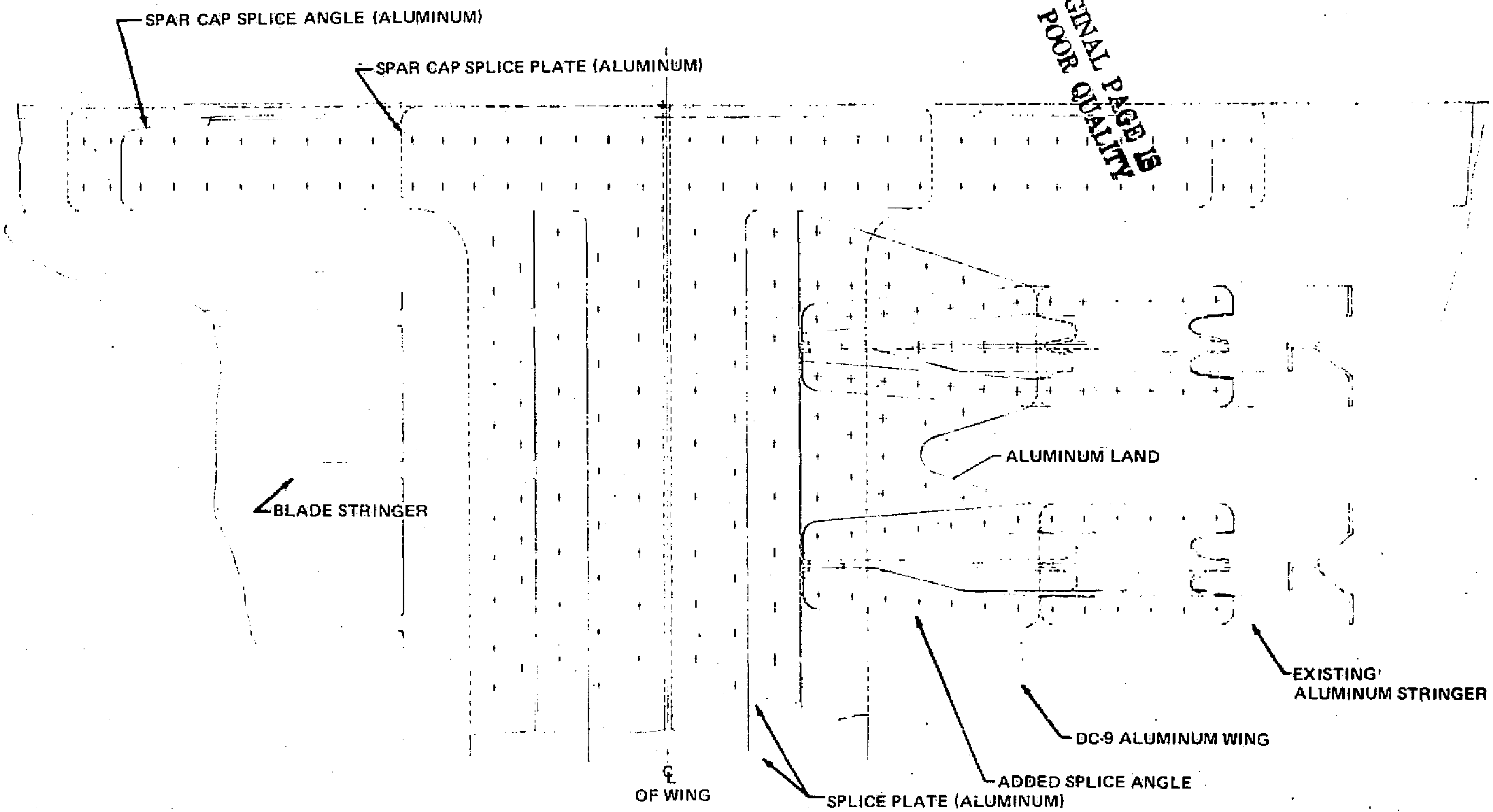


FIGURE 6-22. CENTERLINE WING SPLICE DETAILS



AT-AT

VIEW LOOKING UP AT
INSIDE UPPER SURFACE
OF THE WING
VIEW ROTATED 90 DEG CCW

FIGURE 6-23. LOOKING UP AT UPPER INNER SURFACE OF WING AT CENTERLINE

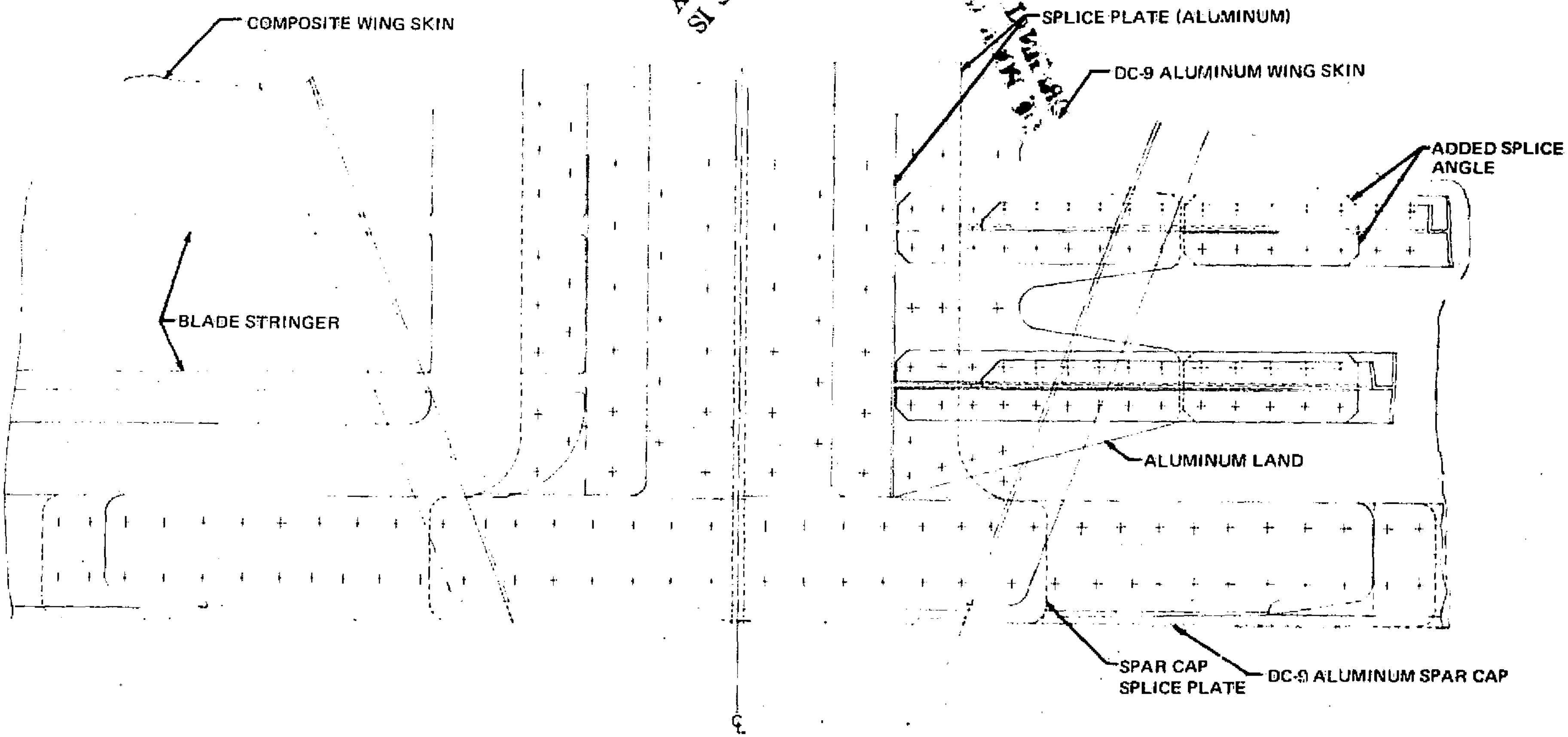
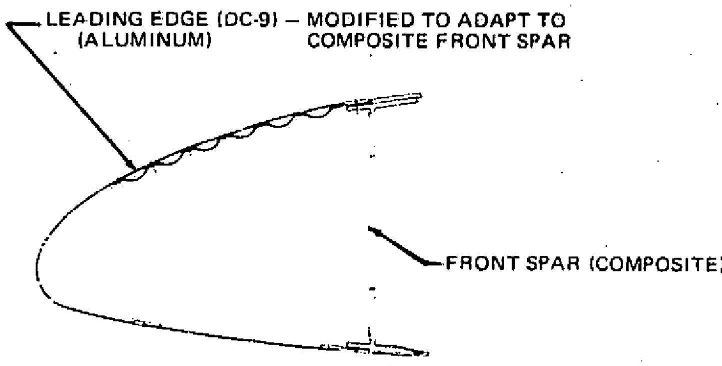
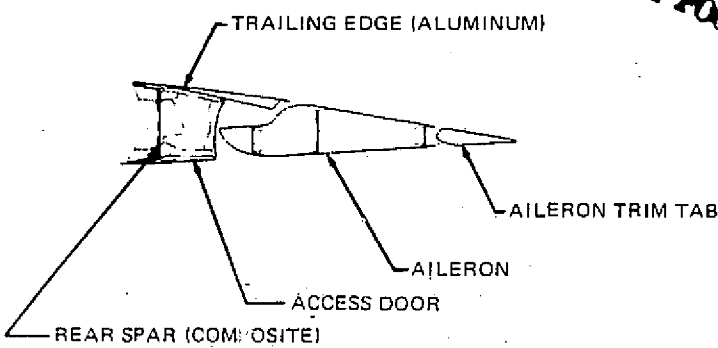


FIGURE 6-24. LOOKING DOWN AT LOWER INNER SURFACE OF WING



WING LEADING EDGE INTERFACE



WING TRAILING EDGE INTERFACE

FIGURE 6-25. LEADING AND TRAILING EDGE INTERFACES

TABLE 6-1
CONCEPTUAL DC-9 CWB WEIGHT SUMMARY

COMPONENT	WEIGHT		WEIGHT SAVINGS		
	METAL DESIGN kg (LB)	CWB DESIGN kg (LB)	kg	(LB)	PERCENT
SKIN PANELS	2090 (4607)	1324 (2920)	765	(1687)	37
SPAR CAPS	518 (1143)	384 (846)	135	(297)	26
SPAR WEBS	202 (445)	159 (351)	43	(94)	21
RIBS AND BHDS	362 (797)	266 (587)	95	(210)	26
CONTINGENCIES	N/A	156 (344)	<156>	(<344>)	–
TOTAL	3172 (6992)	2290 (5048)	882	(1944)	28

Strain (0.003) — 0.003 in./in. maximum permissible strain in any component.

Strength — Component fracture or buckling under ultimate static loads.

This investigation was limited to the weight variations of the cover skin panels. For the purposes of this study, rib and spar weight savings are assumed to be unaffected by the above criteria for all design conditions and the weights shown in Table 6-1 are used for all study cases.

The stiffness critical design used in this comparative study is the conceptual design utilizing blade-stiffened skin panels and shear-resistant spar and rib webs. T-stiffened skin panels with the same spar and rib webs were used for both strain- and strength-constrained designs.

Panel design proceeded along the same lines in each case. Element sizes were computed for both upper and lower skin panels at a number of stations along the span of the wing for DC-9-32 loads and appropriate limitations. The unit panel weights were computed and added to spar, rib, and contingency unit weights to give the total unit wing box weights at each station. These are presented in Figure 6-26 and extended in Figure 6-27, which shows cumulative weight from the centerline of the airplane outboard. A summary of total wing box weights and percentages of aluminum wing box weight and airplane operators empty weight are presented in Table 6-2. The strength criteria ($K_T = 1.0$) is the lower bound for all conditions. The computations of cumulative and total weights are derived from the unit weights of Figure 6-26.

The results show significant weight savings for all design conditions. However, the full strength/weight potential of graphite/epoxy structures is severely limited by the imposition of stiffness and/or strain constraints. Table 6-2 shows the weight savings resulting from stiffness, 0.004 in./in. strain, and the combination of the two as being roughly the same with the 0.004 in./in. strain alone as producing the greatest weight saving. This is 64 percent of the weight reduction provided by a strength-limited design. The smallest saving results from a design constrained by 0.003 in./in. strain and stiffness limitation, which is only 41 percent of the full strength/weight potential. This is a realistic evaluation of the range of weight reduction that can be expected from composite wing design and serves to emphasize the importance of minimizing stiffness and strain constraints.

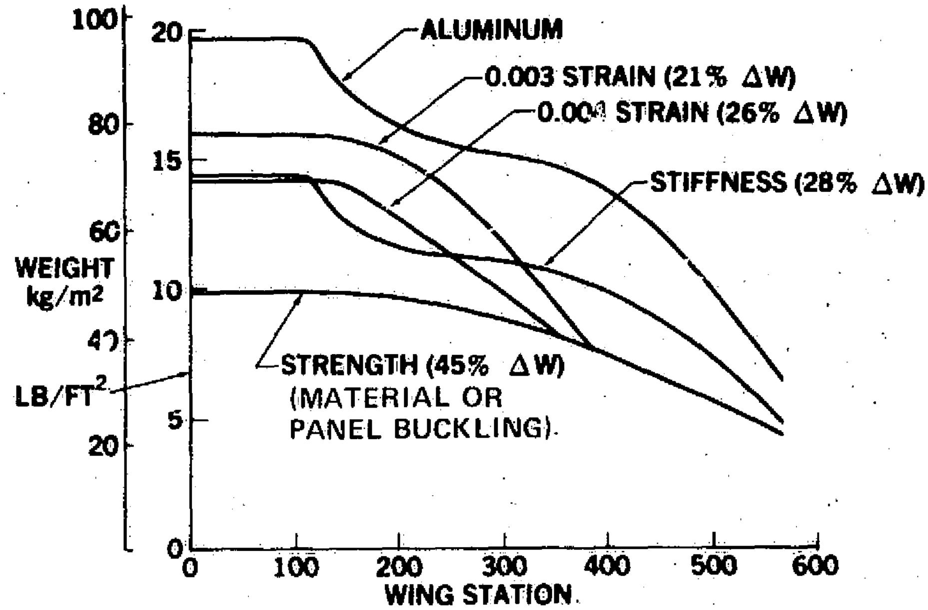


FIGURE 6-26. DC-9 WING BOX WEIGHT DISTRIBUTION

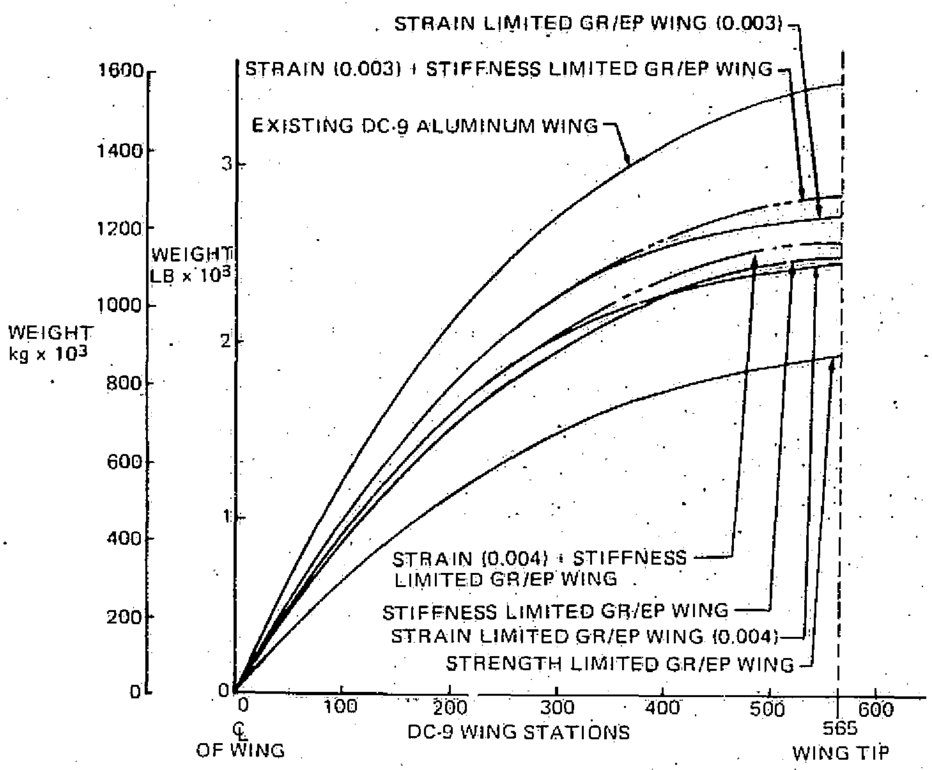


FIGURE 6-27. DC-9 WING BOX ACCUMULATIVE WEIGHT DISTRIBUTION

TABLE 6-2
COMPOSITE DC-9 WING BOX WEIGHT SAVING POTENTIAL

WING BOX DESIGN	WING BOX TOTAL WEIGHT kg (LB)	WING BOX WEIGHT SAVED kg (LB)	WING BOX WEIGHT SAVED (PERCENT)	OEW WEIGHT SAVED (PERCENT)
GR/EP STRENGTH CONSTRAINED	1740 (3835)	1432 (3157)	45	5.36
GR/EP STRAIN 0.004 IN./IN. CONSTRAINED	2250 (4960)	922 (2032)	29	3.45
GR/EP STIFFNESS CONSTRAINED	2290 (5048)	882 (1944)	28	3.30
GR/EP STRAIN 0.004 IN./IN. + STIFFNESS CONSTRAINED	2359 (5200)	813 (1792)	26	3.04
GR/EP STRAIN 0.003 IN./IN. CONSTRAINED	2498 (5508)	673 (1484)	21	2.52
GR/EP STRAIN 0.003 IN./IN. + STIFFNESS CONSTRAINED	2582 (5692	590 (1300)	19	2.21
ALUMINUM	3172 (6992)	-	–	–

SECTION 7
THE DEVELOPMENT PLAN

A composite wing technology program has been defined which will provde the needed technology and data to support the introduction of primary composite wing structure into production aircraft. The parameters upon which the program was constructed, as discussed earlier, include the acceptance factors, the technology assessment, the selection of a DC-9-32 wing for the basic wing configuration, and the selection of program Option 4 to define the details of the plan.

On this basis, a low-cost program has been established which will meet program objectives with an acceptable risk level and will address the issues considered most critical by the commercial air transport community.

The statement of work for the development plan has been sequentially scheduled in six phases, as shown in Figure 7-1. Table 7-1 summarizes the tasks to be accomplished by departmental functions for the six program phases. Cost, schedule, and technical performance can be monitored and evaluated, and program redirection can be effected as downstream developments diverge from predictions. Each phase can be separately funded to allow a reallocation of funds to support the redirection. This will tend to minimize the programmatic risk associated with creative endeavors.

The development plan contains the following provisions, as in Figure 7-2.

- A comprehensive technology development program.
- Design of a DC-9-32 composite wing based on the conceptual design.
- Design and construction of large tools for composite parts.
- Production of flightworthy hardware.
- Test verification to meet FAA structural integrity requirements.
- Installation of a composite wing box on a certified DC-9-32 aircraft with subsequent ground and flight tests to qualify it for commercial revenue service.

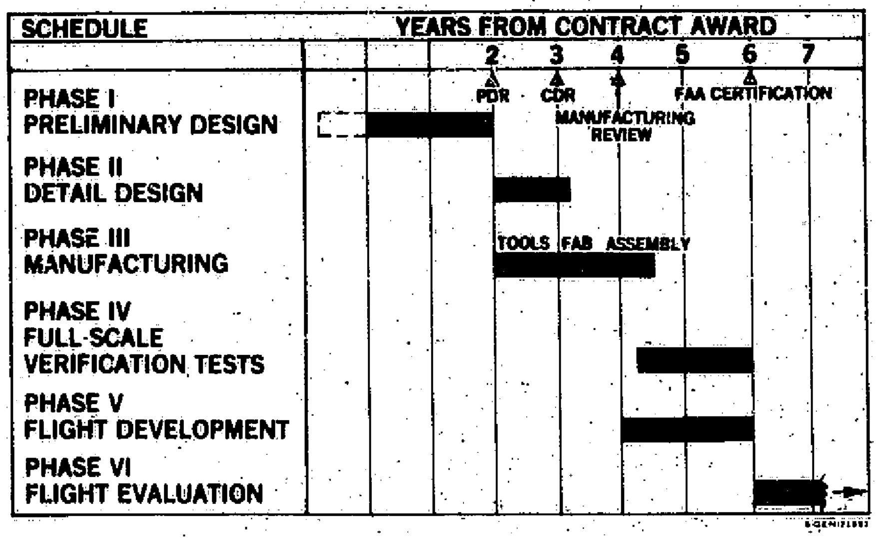


FIGURE 7-1. DEVELOPMENT PLAN SCHEDULE

TABLE 7-1
DEVELOPMENT PLAN

	ENGINEERING PLAN	MATERIAL AND PROCESSES PLAN	MANUFACTURING PLAN	QUALITY ASSURANCE PLAN	TEST PLAN
PHASE I PRELIMINARY DESIGN	DESIGN SYNTHESIS CONCEPT SELECTION	TECHNOLOGY DEVELOPMENT MATERIALS SELECTION	TECHNOLOGY DEVELOPMENT RISK ASSESSMENT	CONCEPT REVIEW	COMPONENT DEVELOPMENT TEST
PHASE II DETAIL DESIGN	FINAL DESIGN AND ANALYSIS PRODUCTION DRAWINGS	MATERIAL AND PRODUCIBILITY STANDARDS	COMPONENT MANUFACTURING VERIFICATION	SPECIFICATIONS PROCEDURES TRAINING	COMPONENT VERIFICATION TESTS
PHASE III MANUFACTURING	SUPPORT DESIGN CHANGES	SUPPORT	1 FULL-SCALE SUBCOMPONENT 3 CWB SEMISPANS	MATERIAL PROCESSES COMPONENTS AND ASSEMBLIES	–
PHASE IV FULL-SCALE VERIFICATION TESTS	TEST REQUIREMENTS SUPPORT DATA ANALYSIS	–	–	–	FULL-SCALE SUBCOMPONENT AND CWB SEMI-SPAN GROUND TESTS
PHASE V FLIGHT DEVELOPMENT	WING INSTALLATION DESIGN TEST REQUIREMENTS DATA ANALYSIS	–	AIRPLANE MODIFICATION 1 CWB SEMISPAN FABRICATION CWB INSTALLATION	–	AIRPLANE GROUND TESTS AND FLIGHT TESTS
PHASE VI FLIGHT EVALUATION	MONITOR EVALUATION	–	–	–	–

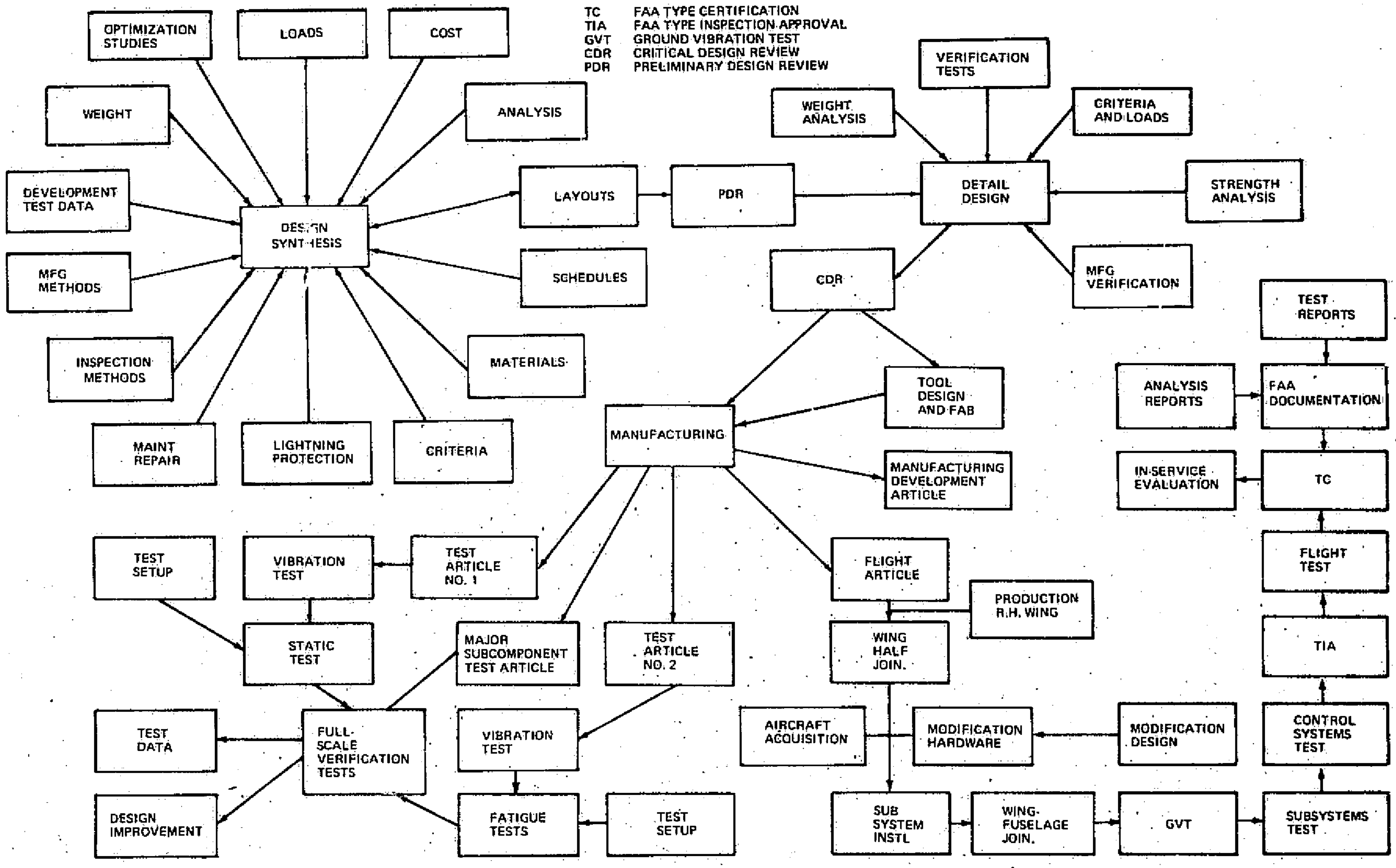


FIGURE 7-2. COMPOSITE WING DEVELOPMENT PLAN

- Monitoring and evaluation of the performance of the composite wing box for 5 years while in revenue service.

The development plan also includes the engineering plan, materials and process plan, manufacturing plan, quality assurance plan, and test plan.

ENGINEERING PLAN

The Engineering Plan consists of the design synthesis in Phase I, detail design in Phase II, engineering support throughout the entire program, and FAA certification.

Design Synthesis

Engineering activity in the design synthesis phase will be devoted to preliminary design, evaluation, and selection of structural concepts for further development. The structural requirements, potential structural concepts, candidate materials, and methods of analysis of composite structures will be brought together in various designs. These designs will form the basis of evaluations in terms of weight, cost, and risk.

The design synthesis process shown in Figure 7-3 is an iterative one which will parallel and interface with the manufacturing, development test, and maintenance and repairability activities. Initial evaluation of structural candidates will determine which concepts are to be designated for development and test. Data from these evaluations will be fed back to the layout effort for design refinement and to the trade studies for reevaluation and elimination of the less efficient concepts. This process will result in the preliminary design of the concept selected for detail design and fabrication of a full-size composite wing box.

Structural Design Criteria and Loads — The basic criterion to be used throughout this program is that the composite wing must be comparable to the aluminum wing in all areas of structural integrity, flight performance, ground handling, and maintenance. To achieve this, the composite wing will be designed to satisfy all Federal Aviation Regulations applicable to the DC-9-32 aircraft. Compliance will be shown in accordance with the guidelines described in Reference 2.

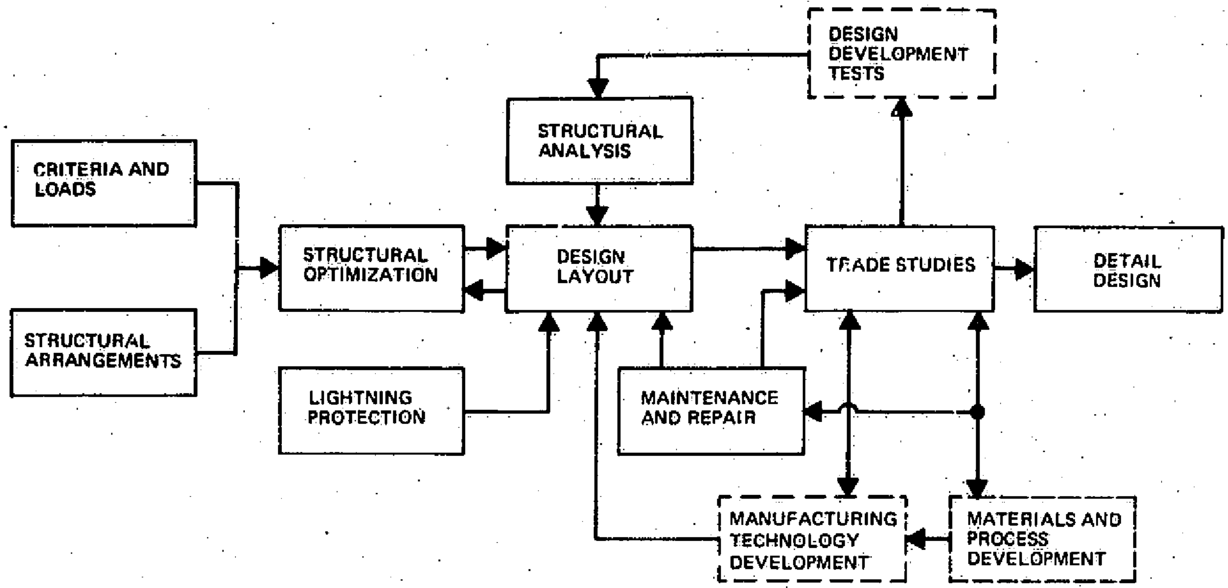


FIGURE 7-3. DESIGN SYNTHESIS PROCESS

Additional design criteria are required to ensure proper installation and performance of a composite wing box on a DC-9 flight article. These include the stiffness criteria requiring the composite box to have the same bending and torsional stiffness as the metal box it replaces and interface criteria which require the locations of all interface structure to remain where they are in existing aircraft. These are the same as applied to the conceptual design, described in Section 6. An FAA criteria summary along with the source of each requirement is presented in Table 7-2.

The loads to be used for composite wing design are the existing DC-9 wing loads. A complete set of these loads was compiled for the conceptual design effort and is presented in Section 6.

Candidate Concepts — Structural concepts are considered in three categories: general arrangements, component concepts, and joints and fittings. General arrangements are the locations of major components such as spars, ribs, and interface structure. Component concepts refer to the various forms of skin panel, spar, and rib construction. The category of joints and fittings includes panel and web joints and fittings required at structural interfaces and system attachment locations.

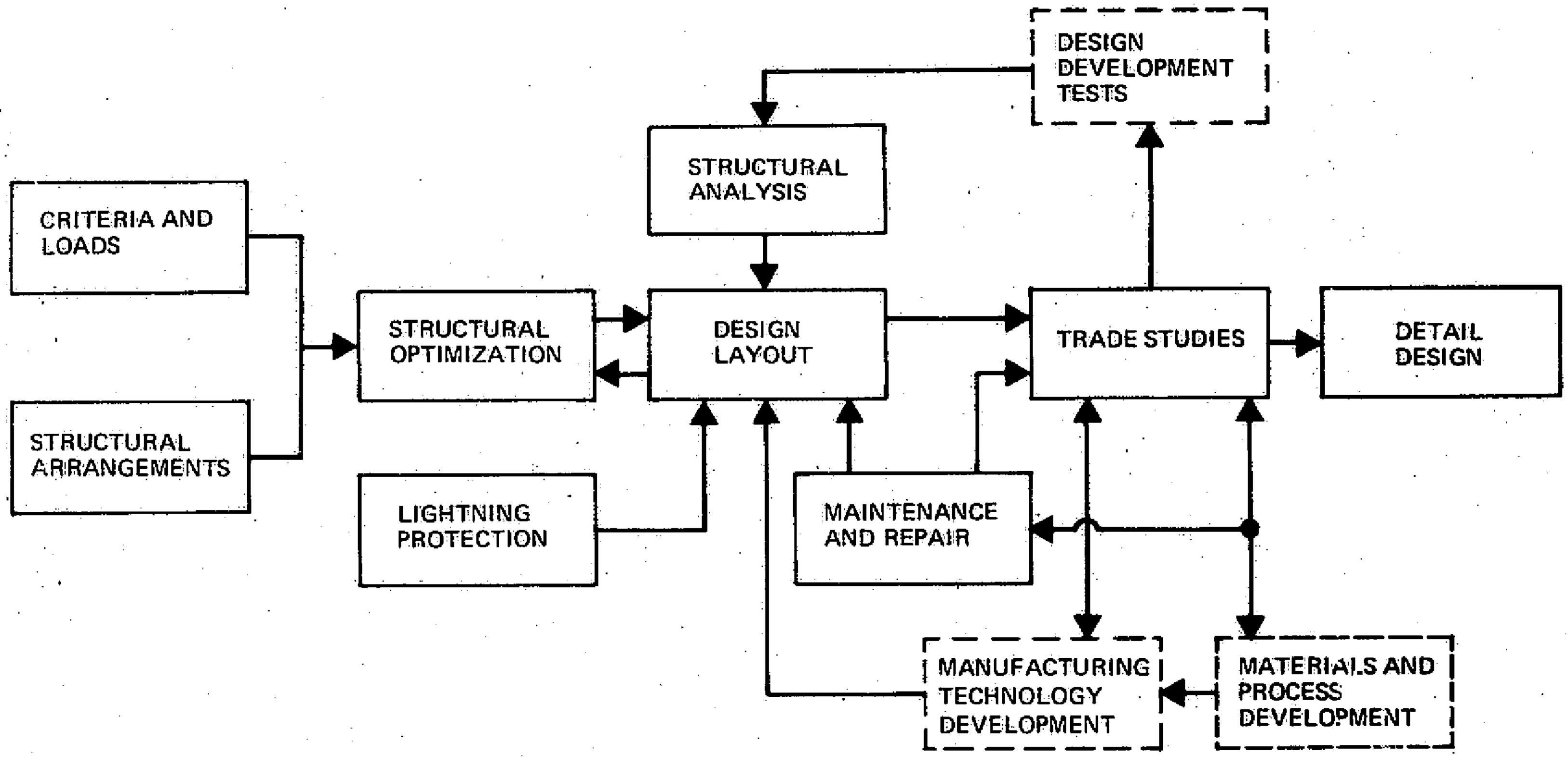

DESIGN DEVELOPMENT TESTS
STRUCTURAL ANALYSIS
CRITERIA AND LOADS
STRUCTURAL OPTIMIZATION
DESIGN LAYOUT
TRADE STUDIES
DETAIL DESIGN
STRUCTURAL ARRANGEMENTS
LIGHTNING PROTECTION
MAINTENANCE AND REPAIR
MANUFACTURING TECHNOLOGY DEVELOPMENT
MATERIALS AND PROCESS DEVELOPMENT

TABLE 7-2.
CERTIFICATION GUIDELINES FOR CIVIL COMPOSITE WING AIRCRAFT STRUCTURES

• MATERIAL ALLOWABLES	– FAR 25.603, 25.613 AND 25.615
• PROOF OF STRUCTURE – STATIC	– FAR 25.305 AND 25.307(a)
• PROOF OF STRUCTURE – FATIGUE/DAMAGE TOLERANCE	– FAR 25.571 (PROPOSED NEW AND APPENDIX)
• CRASHWORTHINESS	– FAR 25.561, 25.721, 25.801(b) (e), AND 25.963(d)
• FLAMMABILITY	– FAR 25.863(b) (5), 25.867, 25.1191 AND 25.1193
• LIGHTNING PROTECTION	– FAR 25.581
• PROTECTION OF STRUCTURE	– FAR 25.609
• QUALITY CONTROL	– FAR 21.143
• REPAIR	– FAR 121.367(a) AND FAR 43.13(a)
• FABRICATION METHODS	– FAR 25.603 AND 25.605

Multirib construction has been selected as the general arrangement for the composite wing box. Other arrangements, including multispar and truss-web, have been considered in previous studies which have verified the multirib concept as the most efficient for most transport wing designs. It is a proven concept and the one chosen for the conceptual design. This is a simple yet versatile approach. The arrangement satisfies all requirements without the use of intersecting internal spars and ribs. This rib orientation permits efficient use of both skin-stringer and sandwich skin panels. The composites will be applied to this concept by direct substitution of composite components for metal ones in the same locations. The composite components will not be identical configurations, but will be designed to utilize the advantages of composite materials most efficiently.

Candidate component concepts, joints, and fittings to be included in the design synthesis are presented in Figure 7-4. Skin panel and web concepts include skin stringer, corrugation, grid-stiffened, and sandwich configurations that have demonstrated efficient application to composite designs in previous efforts. Investigations to date have tended to indicate blade-stiffened panels as the most cost- and weight-effective in this application because of the required stiffness constraints. However, these studies have been preliminary. All concepts presented will be considered candidates until eliminated by a more thorough investigation. Candidate joints and fittings are the standard ones generally considered for composite applications. These include

TYPE	CONFIGURATION	APPLICATION	
		SKIN PANEL	SPAR AND RIB WEBS
FLAT PANEL			
SKIN-STRINGER	BLADE	X	X
	T	X	X
	J	X	X
	HAT	X	X
	HAT	X	X
	HC BLADE	X	
	HC SKN STR	X	
CORRUGATION	SINE WAVE		X
	BEAD		X
GRID STIFFENED	0°-90°	X	
	-45°	X	X
	ISOGRID	X	X
SANDWICH	HC CORE	X	X
	FOAM CORE	X	X

TYPE	CONFIGURATION
JOINTS	INTEGRAL BOND
	MECHANICAL ATTACH
FITTINGS	BONDED TI
	BOLTED AL
	COCURED COMPOSITE

FIGURE 7-4. CANDIDATE COMPONENT CONCEPTS

mechanical attachment either alone or in combination with secondary bonding, cocured titanium fittings, and integral composite fittings. Each will be evaluated for specific applications as required during design.

Structural Optimization — Structural optimization is the initial concept evaluation effort. It serves a twofold purpose. The first is to narrow the field of candidate component concepts to a manageable number for design development. The second is to provide preliminary structural sizing and weight estimates for remaining concepts. The optimization process entails determining the sectional geometry and element sizes which result in the least weight for each candidate. The relative weights of the candidates are then compared and those demonstrating a high degree of structural efficiency without indicating a potential for excessively high cost or risk are retained for further study.

Optimum design studies have been conducted for DC-9 composite wing skin panels during previous in-house programs. The studies were limited to skin-stringer panels and were used for the conceptual design. These studies are applicable to this development program and will be reviewed for completeness and updated and extended as required.

Lightning Protection Features — The nonconductive nature of graphite composite structures relative to aluminum results in potential hazards which require special design considerations. The graphite composite structure design approach will be examined. The associated electrical and instrumentation wiring components will also be reviewed. Critical structure fuel tank and wiring components vulnerable to the adverse lightning and static electricity effects, due to the use of graphite composite structures, will be identified. The lightning and static electricity protection requirements will then be determined.

A tradeoff study will be made to determine the optimum lightning and static electricity protection techniques which will satisfy the design requirements. The study criteria include protection effectiveness, weight, cost, manufacturing ease, and maintainability.

Design Layout — The design layout effort will serve two basic functions. First, it will be used to set geometric constraints on the candidate concepts for optimization studies. Second, it will be used to determine how these concepts can be incorporated into an integrated wing box design and what the penalties will be.

Preliminary layouts of the structural candidates will be generated in enough detail to determine limitations on element size and spacing for optimization studies. These will include advanced design of joints between skin panels, spars, and ribs, rough layouts of interfacing systems, and laminate patterns in areas of low loading. Particular emphasis will be placed on the fuel system requirements which may affect the stringer spacing on skin-stringer type panels. The limitations required by manufacturing and service considerations will be defined and incorporated.

Advanced design layouts of those concepts selected as a result of the optimization studies will be developed. These layouts will define the major structural and manufacturing aspects of concepts integration into a complete wing box structure. This effort will proceed along the same lines as described for the conceptual design. Layouts will be made of major structural members and typical substructure, joints, and interface structure, as shown in Figure 7-5. The basic sections of the skin panels and spar webs will be designed at a number of stations along the wing span to determine how they can be tapered for minimum weight. Typical panel-to-spar and panel-to-rib

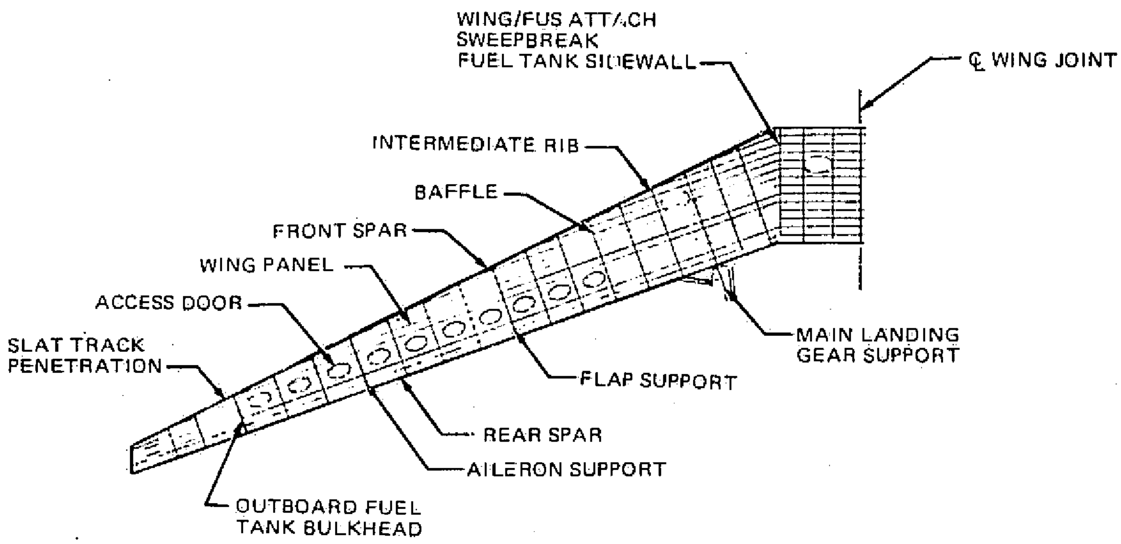


FIGURE 7-5. PRELIMINARY DESIGN LAYOUT SCOPE

joints will be laid out. The internal substructure will be defined by preliminary layouts of a typical support rib, one fuel baffle, and the outboard tank bulkhead. Interface supporting ribs and fittings will be designed at the wing centerline joint, the wing-to-fuselage attachment, the landing gear support, one flap support, one aileron support, and one slat track support and penetration.

Advanced design layout of this scope will adequately define each candidate to the degree required for trade study evaluation and determination of development test requirements and specimen design. The layouts will be continually updated as more complete strength analyses refine component sizing, and manufacturing, and maintenance, and test data indicate required design changes. The changing layouts will be continually reevaluated in trade studies as the design is synthesized.

Trade Studies — The trade studies will be the second evaluation effort after structural optimization. These studies will compare the candidate concepts as defined by design layout in terms of weight, cost, and risk. The result is the selection of the concept designated for detail design and fabrication.

Initial trade studies will narrow the field of candidates down to a number that can reasonably be carried thoroughly through the development and test efforts while permitting the program to remain within budget. The general arrangement concepts will be limited to one, that of multiweb construction. It is intended that skin panel, spar web, and rib candidates be narrowed down to two or possibly three by initial trades. It is doubtful that any of the joint or fitting concepts included in Figure 7-4 will be eliminated without the benefit of manufacturing and test data.

The trade studies will keep abreast of all development efforts. As the layouts are revised by application and analysis of new data, the trade studies will be updated. Candidates will be eliminated as deficiencies are established until one concept is clearly established as the most efficient, considering all areas of design, fabrication, maintenance, and repair.

Structural Analysis — The structural analysis effort entails methods development and structural sizing. The approach includes theoretical analysis and the definition of development test plan requirements and interpretation of results.

Plate and shell analysis methods are used in the design synthesis phase. This is primarily a preliminary design and evaluation effort. Composite structural analysis is based on orthotropic analysis techniques which have been developed at Douglas during the past few years on both in-house and contracted programs. Both design charts and computer programs are available for composite structural analysis, but the computer programs are the most versatile and generally provide the most complete analysis. Programs which can be used to optimize and analyze basic components are presented in Table 7-3. Programs available for analysis of the types of joints and fittings applicable to wing box design are presented in Table 7-4.

Blank boxes in Tables 7-3 and 7-4 indicate that no computer program (or design chart) is available at this time for the specified structure and loading condition. The analytical approach to development of missing programs is known. Only time and effort are required to complete all those required for wing box structural analysis.

The strength of skin panels, spars, and ribs under basic wing bending, shear, and torque will be considered in the structural optimization. Additional strength analysis of these components will include critical combinations of loadings as well as internal fuel pressure. All modes of failure will be investigated, including stability and fracture resulting from critical combinations of tension, compression, and shear.

Special attention will be given to joints, fittings, and supporting structure. Analysis of strength of mechanical attachments and local areas in the vicinity of fittings will require analysis of stress distribution, theoretical strength prediction, and interpretation of data as they become available from development tests.

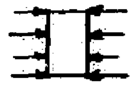
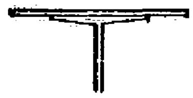
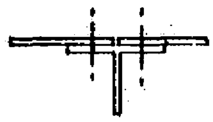
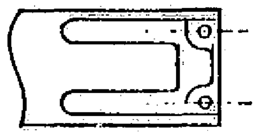
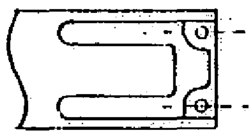
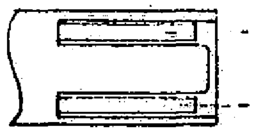
TABLE 7-3
AVAILABLE COMPUTER PROGRAMS
BASIC COMPONENTS

TYPE	CONFIGURATION	APPLICATION		LOADING CONDITION		
		SKIN PANEL	SPAR AND RIB WEBS	AXIAL	SHEAR	NORMAL PRESSURE
FLAT PANEL				PSB PCB PSFB	PSB PCB	PSI PSPI
SKIN-STRINGER	BLADE	X	X	BSC	BSC	(1)
	T	X	X	JSC	JSC	(1)
	J	X	X	JSC	JSC	(1)
	HAT	X	X	HSC	HSC	(1)
	HAT	X	X			
	HC BLADE	X				
	HC SKN STR	X				
CORRUGATION	SINE WAVE		X	NA		
	BEAD		X	NA		
GRID STIFFENED	0°-90°	X		WSC	WSC	
	±45°	X	X			
	ISOGRID	X	X			
SANDWICH	HC CORE	X	X	SPSB SPCB	SPSB SPCB	
	FOAM CORE	X	X	SPSB SPCB	SPSB SPCB	

(1) PROGRAM FOR SIMPLY SUPPORTED CASE ONLY

NA NOT APPLICABLE

TABLE 7-4
AVAILABLE COMPUTER PROGRAMS
JOINTS AND FITTINGS

TYPE	CONFIGURATION		LOADING CONDITION	
			(1) PANEL 	SHEAR 
JOINTS	INTEGRAL BOND		DBLJT SCARF	
	MECHANICAL ATTACH		(2) C6SY-1	
FITTINGS	BONDED TI		DBLJT SCARF	
	BOLTED AL		C6SY-2	
	COCURED COMPOSITE		DBLJT SCARF	

(1) FOR DOUBLE LAP, SCARFED, AND SUPPORTED SINGLE-LAP JOINTS

(2) FOR PSEUDO ISOTROPIC LAMINATE JOINTS ONLY

Structural analysis of components, joints, fittings, and assemblies will include assessment of damage tolerance, durability, and repair procedures. These analyses will be largely based on interpretation of test data.

Structural analysis results will be applied to the design layouts in the form of refinements to structural sizing.

Detail Design

The final design of the ground and flight test articles will be accomplished in Phase II. The design of the concept selected as most efficient in the preliminary design phase will be finalized. Each component of the structure must be designed in detail and analyzed for all critical loading conditions. All aspects of the structure will be considered. Structural concepts determined as the best approaches and designed for representative applications during preliminary design will be designed in detail for all applications. Design layouts will be made to include all major components, joints, interface structure, and system provisions. Detail drawings will be made to permit fabrication of verification test specimens and complete wing box structure. Verification test requirements will be defined.

Design Layouts — The final detail design layout effort will consist of updating and extending the preliminary layout of the concept selected for detail design in the preliminary design phase. The layouts will define component geometry including planform and element spacing as well as element sizes. The sizes of composite elements such as skins and stiffeners will be defined by a number of plies and orientation of each as well as dimensions. Provisions for cutouts, integral stiffening, joints, and any other special features will be defined by detail layout of laminate patterns in each area involved. The sizing will be based on detail structural analysis and manufacturing cost considerations.

Detail Drawings — Detail drawings will be made of the designs defined by layout. Prototype drawing of structural components, assemblies, and installations will be made in the detail required for fabrication of test specimens and full-size wing box structural assemblies. The drawings will take a form similar to drawings of metal parts with definitions added on the number of plies, their orientation, and location.

Structural Analysis — Strength analysis of final detail composite design will be based on finite-element analysis. Internal loads included in skin panels, spar webs, and ribs as well as local stress distribution in areas of high concentrated loads such as wing-to-fuselage joint and landing gear support will be calculated by in-house computer programs. Computer graphics are used to assist input and output while finite-element programs generate internal loads. Initial structural modeling will be done with the Douglas Computer Graphics Structural Analysis (CGSM) program which interfaces directly with both NASTRAN and Computer-Aided Structural Design (CASD), a Douglas program. Either of these programs can be used to generate finite-element analysis. The output is presented graphically as internal loads superimposed on the structural model. A hard copy of the output can be obtained by a Gerber plot.

The analysis of the strength of the structural members under the influence of these internal loads as well as damage tolerance and durability analysis is performed in the same way as the preliminary structural analysis. Assessment of crashworthiness provisions will be included and will entail theoretical analysis and interpretation of test results.

Lightning and Static Electricity Protection — The overall design will be determined for protection of the composite wing from lightning and static electricity. The manufacturing and processing methods for incorporating the protection techniques will be established.

The existing electrical subsystems and the associated wiring installations will be retained in the composite wing design. Additional bonding, grounding, and transient suppression/filter devices will be incorporated as required.

Weight Analysis — The total weight of the composite component will be estimated. All weights will be updated to incorporate revisions as they are released. Reports will be published comparing the composite design to the metal design at the lowest practicable level of detail. A target weight for the composite component will be established. The current weight of the component will be monitored continuously and solutions to any overweight problems will be discussed with the designers and management. Fabricated parts of the composite wing component will be weighed. The calculated weights will be compared with actual weights and any discrepancies will be reconciled.

<u>Sustaining Engineering</u>

Engineering design and analysis support will be provided throughout the program. Design changes and rework drawings will be provided as required during the Phase III manufacturing effort. Ground test requirements for the composite wing box will be defined and support provided for setup, design, and construction of tests during Phase IV. The tests will be monitored and results interpreted. Modification drawings will be prepared in Phase V to permit the adaptation of the composite wing box to a DC-9 flight article. Aircraft ground vibration test and flight tests will be defined, monitored, and results evaluated. The Phase VI in-service flight program will be defined and reviewed at designated periods.

FAA Certification

A comprehensive FAA certification plan will be prepared in Phase I. Results of design parameters analysis and test data will be compiled in reports designated by the plan as they become available and submitted for FAA approval. These reports will include design criteria, external loads, and material properties compiled in Phase I; internal loads, stress analysis, fatigue/damage tolerance analysis, and component verification test results of Phase II; ground tests of Phase IV; and flight tests of Phase V.

MATERIALS AND PROCESS PLAN

The Materials and Producibility Engineering department will support the Engineering Design section during the preliminary design (Phase I) and detail design (Phase II). This support will include the selection of materials, the environmental conditions, and the assessment of the manufacturing ease or producibility of the design.

The material systems to be used for the wing structure will be selected at the time of the actual program. The materials selected will have proven handling and processing characteristics and acceptable mechanical and environmental properties, resistance to microbiological contamination of fuel, and impact toughness.

Design data specimens will be fabricated, conditioned, and tested as prescribed by Structural Engineering, using manufacturing techniques proposed for fabrication of the large wing structure (time, temperature, pressure, and methods).

Phase I

Technology development by Materials and Producibility Engineering is recommended in Phase I for two disciplines — nondestructive testing and long-term contaminated fuel environment — as discussed in the following text.

Nondestructive Testing

Resin Content Measurement — Ultrasonic velocity variations and neutron gauging techniques appear as viable methods for quantitatively measuring resin content in graphite/epoxy composite structures. Panels containing variations in resin content will be fabricated, analyzed for resin content by nondestructive testing techniques, and checked for resin content by chemical digestion as reference. The panels will be cut and tested for flexural strength and short beam shear strength to verify their mechanical quality. An analysis will be conducted to correlate the relationships for nondestructive testing to measure and establish the laminate resin content.

Void Content Measurement — Ultrasonic attenuation appears to be a viable method of quantitatively measuring void content. Studies will be made on typical thickness graphite/epoxy composite laminate specimens to determine the optimum ultrasonic test frequency, test methods (e.g., pulse-echo or through-transmission), and search-unit size. Various void content reference standards will be fabricated and tested to arrive at a relationship between void content and ultrasonic attenuation. All specimens will be mechanically tested to establish the relationship between void content and strength.

In-Service Aging and Environmental Effects — Boeing is working on a program to determine the environmental effects on graphite/epoxy composite structures. However, no nondestructive testing method of evaluation was included in this program; only mechanical tests are performed periodically. If graphite/epoxy composite structures are to be used on primary structure for commercial aircraft, nondestructive testing methods will be required to determine the degradation of the structure as related to strength and fatigue life. Development of a quick, low-cost, and reliable nondestructive testing technique to determine a change in structural characteristics will be a goal of this program.

Fire Damage – Aircraft structure, especially wing structure, is subject to fire damage in the area of the wheel well. The significance of fire damage must be determined. Nondestructive testing methods will be evaluated to determine if they can relate the fire damage to loss in physical properties of fatigue life. Similar relationships have been established between eddy-current conductivity, hardness, and loss in yield or ultimate strength. This program will consist of fabrication of composite panels, nondestructive testing control tests, exposure to controlled fire environment, determination of extent of damage area by nondestructive testing, and final testing for retained mechanical strength properties.

Effects of Defects Determination – A relationship will be established between the frequency/severity of defects, such as interply porosity, delaminations, voids, and resin and void (porosity) content, and the strength and durability of the graphite/epoxy composite structure for the wing program. Specimens of flat configuration and later specimens of wing structure configuration will be fabricated with various defects intentionally included. These defects will be located and measured by nondestructive testing methods. The specimens will be fatigue-tested and flaw growth will be monitored as a function of applied load cycles. The objective is to establish nondestructive testing standards for the size and location of the critical defects that have a significant effect on the durability of a composite wing structure.

Long-Term Fuel Environment

The resistance of composite materials to a long-term contaminated fuel environment must be proven. Graphite/epoxy composite specimens, both uncoated and coated, will be fabricated with polyurethane fuel tank coating (MIL-C-27225) and immersed in a kerosene fuel/mineral salt water environment for 6 months to a year. These tests will immerse specimens in a sterile kerosene/mineral salt control, using test media inoculated with microbial-contaminated fuel with and without fuel biocides added periodically, and running other tests where selected organic acids are added. At the end of the exposure period the specimens will be examined visually and microscopically; weight, volume, and electrical resistance changes measured; and the specimens then tested for changes in mechanical properties (flexural strength and modulus and horizontal shear).

Phases II and III

A materials specification will be prepared in Phase II to identify the basic material handling, physical, and composite laminate structural material properties. The specification will document purchasing instructions, quality control test procedures for incoming material, and acceptance requirements, storage conditions, and requalification procedures for material B-stage and cured laminate.

A processing standard will be prepared that will prescribe the materials and the detailed, step-by-step manufacturing process for the wing structure. The processing standard will include direction for quality assurance provisions and acceptance/rejection requirements and procedures.

A nondestructive test specification will be prepared to prescribe the detail nondestructive testing methods and acceptance criteria to be used for the wing structure.

Materials and Producibility Engineering will assist and support Manufacturing during fabrication of the Phase II subcomponents and Phase III wing structure sections. Their efforts will include surveillance of manufacturing operations, procedural techniques, quality control and inspection records, and participation in any rework that may be necessary.

MANUFACTURING PLAN

The prospective manufacturing problems associated with producing a composite wing, as discussed in the technology assessment, are based upon the experience we have gained so far. In any major program which extends technical capabilities, more questions arise during the development effort than were anticipated. A study program can only discuss the predictable problems and propose paths for solutions. An innovative program that extends the limits of existing technology requires more development to support the advanced work. Our concepts for producing composite wings for a production line are intended to eventually make the cost of a composite wing equal to or less than the cost of an aluminum wing. The scope of a wing program should not be limited to satisfying the immediate need for producing one part, but rather to open up the potential applications of composites so that the full advantages of integrally cured products can be realized.

Phase I — Technology Development

In order to determine the manufacturing costs, identify the technical problems, and select the most feasible method of molding a full-scale wing, subscale tests must be conducted to supply data. This is a basic requirement if any degree of integral curing will be propsed for the flight articles. Even for a less sophisticated approach, with mechanically joined components, building a series of subscale boxes will generate experience and reliability in less elapsed time and with a lower risk of loss than would be the case with a full wing section.

Through a series of increasingly sophisticated manufacturing trials on graphite wing box sections, alternatives can be evaluated with reliable, realistic data.

It is recommended that a subscale box approximately 8.5 meters (28 feet) long and 1.5 meters (5 feet) wide be constructed as a test case for comparing various manufacturing procedures, Figure 7-6. This box will fit into our existing autoclave, and represents a half-sized version of a DC-9 wing. The box is large enough to demonstrate and explore the manufacturing details, but not so massive as to require special facilities.

The choice of tooling design, materials, and fabrication methods must be explored to select a combination that will maintain dimensional control of the part, not introduce cooldown stresses on the cured part, avoid expensive machining processes, minimize thermal expansion mismatch, be easy to handle, and be cost-effective.

Figure 7-7 represents a tool structure used on the PABST program for large, metal-bonded, curved panels. An egg-crate substructure was fitted with adjustable stud attachments which permitted a lofted surface to be rough-formed and set to proper contour by locally adjusting the height of studs supporting the surface. Typically, the lofted surface was aluminum.

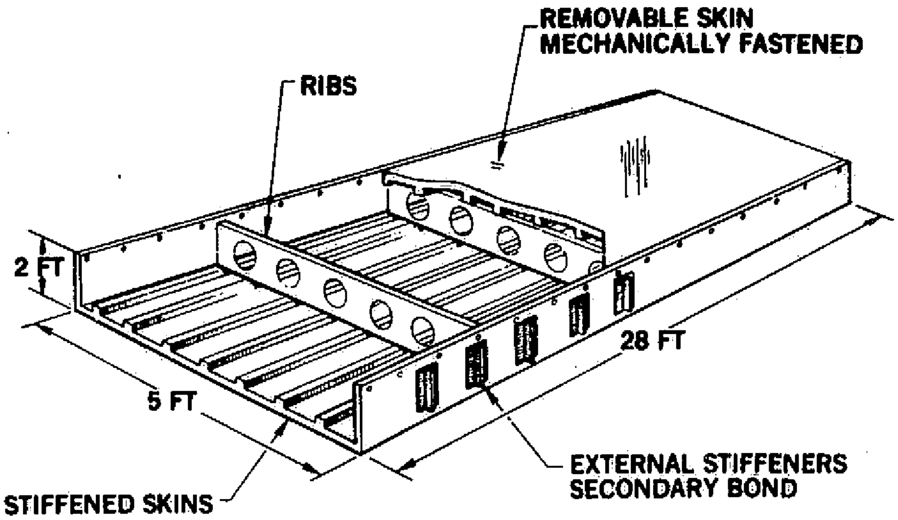


FIGURE 7-6. SUBSCALE COMPOSITE WING BOX DESIGNED TO VERIFY TOOLING/MANUFACTURING FEASIBILITY

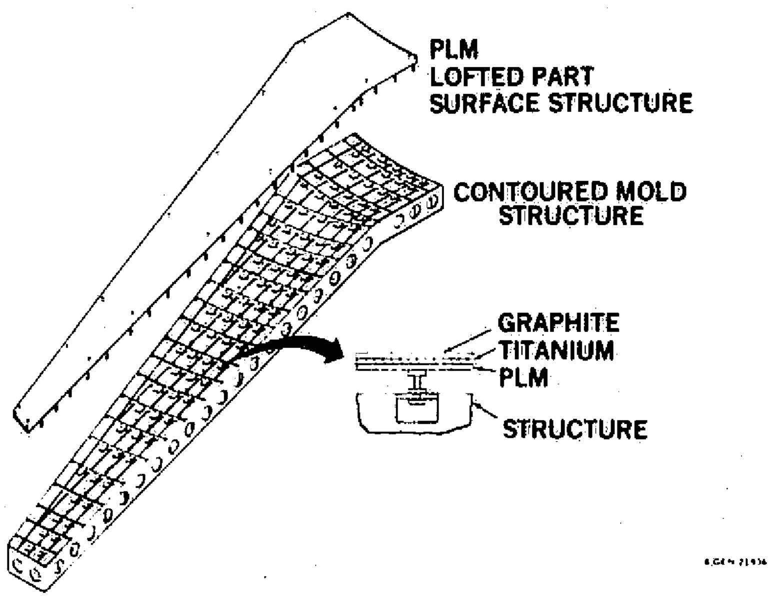


FIGURE 7-7. PLASTIC LAMINATING MOLD (PLM)

For molding graphite composite wings, a caul sheet of titanium placed on the aluminum will reduce the thermal mismatch. The subscale test box will be fabricated using a similar tool to verify the accuracy of the final part to the drawing. It will also enable tooling corrections to be made to compensate for thermal effects.

The final choice of a fabrication method for the full-scale composite box will depend upon the most economical and reliable method demonstrated on the subscale box.

Three methods of construction are proposed for evaluation:

1. Conventional-Bolted

   Stiffened skins are mechanically joined to individual front and rear spars and ribs are secured to spars with fasteners (Figure 7-8). This represents the conventional approach to wing construction. No significant advances in integral curing technology will be obtained by this approach; however, it is lowest risk in terms of potential material loss.

2. Egg-Crate

   The egg-crate approach, where front and rear spars and ribs are cocured (Figure 7-9). Stiffened skins are separately cured, then fastened to the egg-crate substructure. Several molding concepts can be used to cure the spar/rib structure such as inflatable mandrels or trapped rubber.

3. Integrally Cured

   An integrally cured box uses a combination of inflatable mandrels and trapped rubber similar to the box developed in a Company-sponsored program (Figure 7-10). This concept represents the most radical approach with the greatest chance for cost reduction and also the greatest risk.

Each of these methods will be carefully monitored for tooling and manufacturing cost data to permit valid comparisons to be made of the process on an economic basis.

An assessment of each assembly method was made for the risk of failure during the cure cycle. Table 7-5 compares each method, showing the corresponding molding process and the manufacturing risk expected.

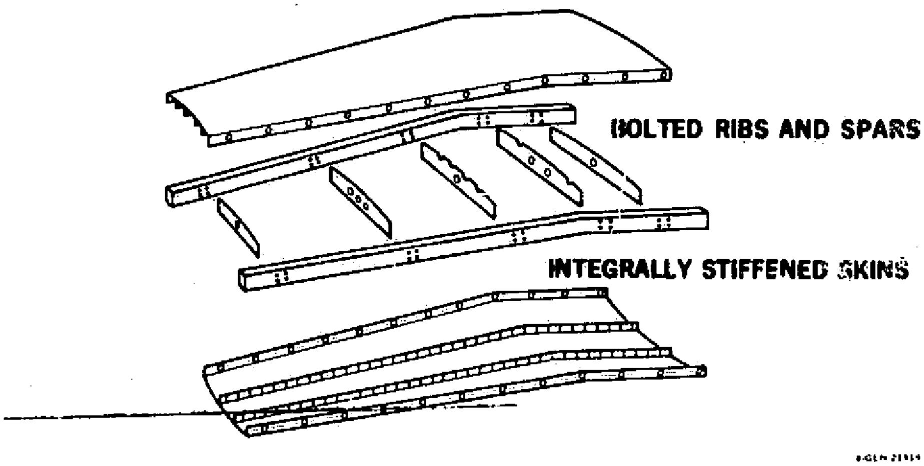


FIGURE 7-8. CONVENTIONAL-BOLTED CONCEPT

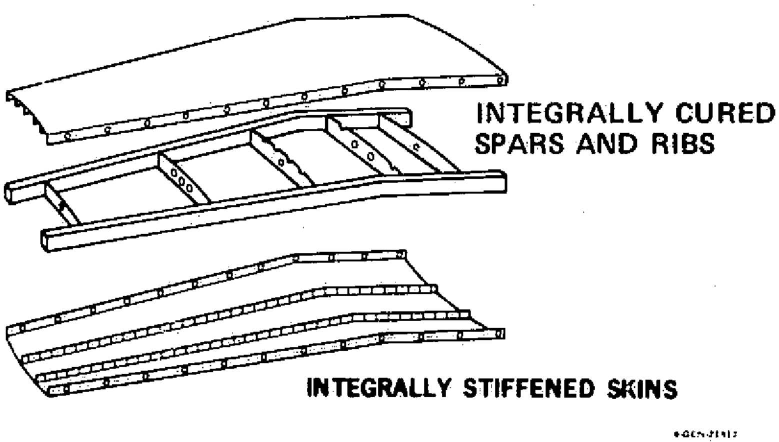


FIGURE 7-9. EGG-CRATE CONCEPT

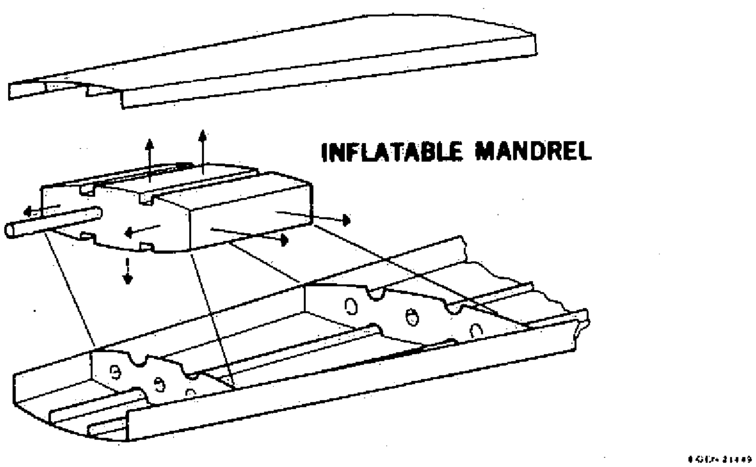


FIGURE 7-10. INTEGRALLY CURED CONCEPT

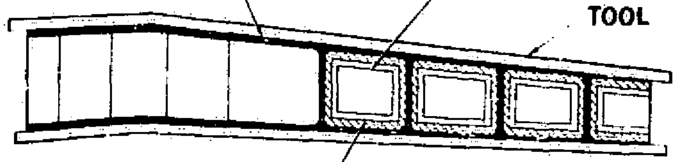
TABLE 7-5
RISK ASSESSMENT

ASSEMBLY METHOD	MOLDING PROCESS	MANUFACTURING RISK
CONVENTIONAL	AUTOCLAVE	RELATIVELY LOW — CURRENT TECHNOLOGY
EGG-CRATE	TRAPPED RUBBER AND AUTOCLAVE OR OVEN	MODERATE — UNIFORM PRESSURE CONTROL DETAIL LOCATION BAG FAILURE
INTEGRALLY CURED	INFLATABLE MANDRELS AND AUTOCLAVE	HIGH — INFLATABLE LEAKAGE LAYUP ON MANDRELS AUTOCLAVE FAILURE

The conventional assembly method, using standard autoclave cure, is the baseline approach. As part sizes increase to span lengths of 15.25 meters (50 feet), some additional risk is introduced because of the increased possibility of vacuum bag failure. The main drawback of the conventional method is the lack of integral construction with this process. All substructural components, spars, and ribs are individually cured and assembled with mechanical fasteners.

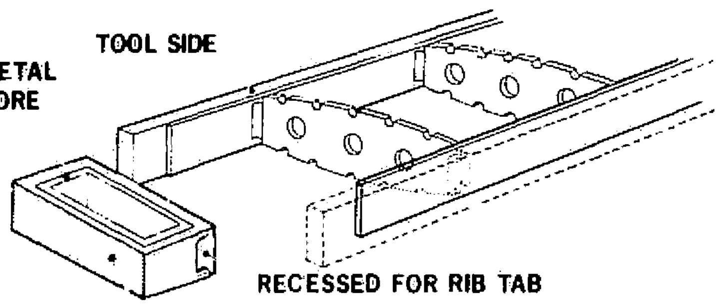
By using the cocured substructure approach (egg-crate) in combination with separately cured stiffened skins, the spars and ribs can be produced without need for fasteners. Further, the fit-up of ribs to the spars will automatically be a net fit. This eliminates the requirement for liquid shimming and tedious inspection efforts to verify gap tolerance. It also reduces the potential fuel leak paths around fasteners by eliminating hundreds of fasteners.

The risk associated with the egg-crate method is judged to be moderate because of the requirement to hold the part location, the need for uniform pressure, and the possibility of bag failure if the autoclave is used for final cure. A method for reducing the risk is to avoid the autoclave by using the trapped rubber process. Figure 7-11 shows a method of curing with rubberfaced inserts that fit between each rib. An isometric view of this concept is presented in Figure 7-12. The silicone rubber expands to cure the ribs and spars together.



OVEN CURE OR AUTOCLAVE CURE

FIGURE 7-11. EGG-CRATE TOOLING



SILICONE RUBBER FACES

FIGURE 7-12. MEDIUM-RISK CONCEPT

The assembly is cured in an oven without external pressure. Tooling can be fabricated from aluminum to increase thermal conductivity and reduce heating time.

The most innovative concept for molding composite wings is to integrally cure the stiffened skins, spars, and possibly the ribs in one cure cycle. This approach would drastically reduce the work now required to assemble the wing by eliminating most of the mechanical attachments. The drilling, countersinking, fastener installation, inspection, record-keeping, and fit-up problems that are typical for metal wings would be reduced, with a corresponding reduction in manufacturing costs.

Douglas has demonstrated that small wing box sections can be fabricated by the use of inflatable mandrels and trapped rubber. However, scale-up from a 2-meter (7-foot) box to a full-sized 15-meter (50-foot) wing represents a very substantial increase in complexity and risk of loss during cure with today's manufacturing technology. In order to improve the reliability of the process and decrease the risk, development work must be continued toward perfecting the construction and longevity of inflatable mandrels. Monolithic curing of the full wing semispan poses too great a risk at the present, but it is conceivable that such a manufacturing approach may become feasible in the future.

Phase II — Production Readiness

A variety of composite specimens and test parts will be fabricated to verify design and to provide data on strength, joint characteristics, fasteners, etc. The manufacturing composites center will produce these specimens and parts as required. Autoclave facilities, machining equipment, and technicians familiar with graphite/epoxy laminates will be assigned to this area. In addition, specialized tooling support from manufacturing research engineers will expedite test parts on a quick-response basis without the need for formal tool designs. Other parts will be fabricated beyond those required for the test program if needed to verify manufacturing methods and processes.

Where feasible, large test parts can be used for verification of the fabrication processes where planning papers, procedures, and quality assurance methods can be demonstrated. The usual problem areas that arise in support of building first article parts can be resolved early to reduce the impact on the full-scale composite wing box.

Phase III — Manufacturing

Phase III covers the technology, processes, and other tasks required to produce graphite/epoxy composite full-scale wing structures for commercial aircraft.

In Phase III, Manufacturing will produce three full-scale, left-hand DC-9 wing boxes and one major subcomponent for laboratory test. Advanced techniques in composite application will be utilized, along with internally applied pressure curing techniques.

The program will be directed toward techniques for rapid and repetitive layup of composite preimpregnated materials. Extensive utilization of automated and memory-controlled tooling is planned for tape and broad goods.

Facilities will be provided for the composite wing production, including receiving, layup, cold storage, curing inspection, trimming, and subassembly. Fabrication will proceed as though in a production mode with all associated planning documentation, proof of compliance with drawings, personnel training and certification, inspection criteria, and facility development.

Major emphasis will be placed on curing large assemblies as integrated units made up of skins and associated structural support elements. Assembly of major composite elements will require minimal mechanical fastening while maximizing the new technology and structural integrity of composite integrated assemblies.

Manufacturing Approach — The DC-9 composite wing is to be produced as a four-element assembly. The skins will be formed with integral stiffening blades; ribs will be individual parts mechanically fastened to the skin structure; and spars will be fabricated separately and mechanically attached to the skin closeout structure. This approach is compatible with the conceptual design defined earlier.

Manufacture of the composite wing is to be undertaken on an individual basis; however, the methods of manufacture will be oriented toward large production runs typical of commercial manufacturing operations.

Composite structures will be produced utilizing manufacturing technology obtained from the DC-10 upper aft rudder and the DC-10 vertical stabilizer programs.

Production and manufacturing cost-estimating systems and records will be used to generate costs for tooling, composites, fabrication, assembly, and metal components.

Planning will utilize a low-cost, one-of-a-kind producibility approach for hardware generation, but will provide data for subsequent input to production costs.

The planning control center will be responsible for releasing documents to the manufacturing R&D center, quick-response tooling aids, and machine shops.

Release planning will utilize existing DC-9 document control procedures. Fabrication orders will be processed through production control for work assignment and status with respect to the schedule.

Large skin tools will be rolled from metal, with final contours N/C machined. The surface will be supported on aluminum egg-crate supports with stud-welded attachments. Titanium caul sheets will be used to minimize thermal expansion mismatch between the graphite skins and the tool surface. Forming tools will be designed for oven and autoclave usage with provisions for bagging, sensor application, and a temperature rise capability of 2.25°C per minute (4°F per minute), whether externally applied or boosted through use of cal-rod heating elements buried in tool components.

Assembly tooling will be for a limited production run, but of the type that can be converted to long-run, permanent tools. Master tools will be confined to existing tooling for provision of hingeline and contours and for overall dimensioning of the wing box. No new tools are anticipated.

Tools will be designed to compensate for thermal expansion during the curing cycles.

Assembly Plans — The detail parts and large stiffened skins will be fabricated in the composite manufacturing facility shown in Figure 7-13. This area is dedicated to preparation, layup, and curing of composite aircraft parts. Metal details such as the titanium doubler and aluminum bulkheads will be fabricated in the normal production shops. Other parts and assemblies common to conventional DC-9 aircraft wings will be available through normal production control groups.

Parts and assemblies will be sent to the composite center for final assembly and acceptance of completed wing box units. Subassemblies will be joined

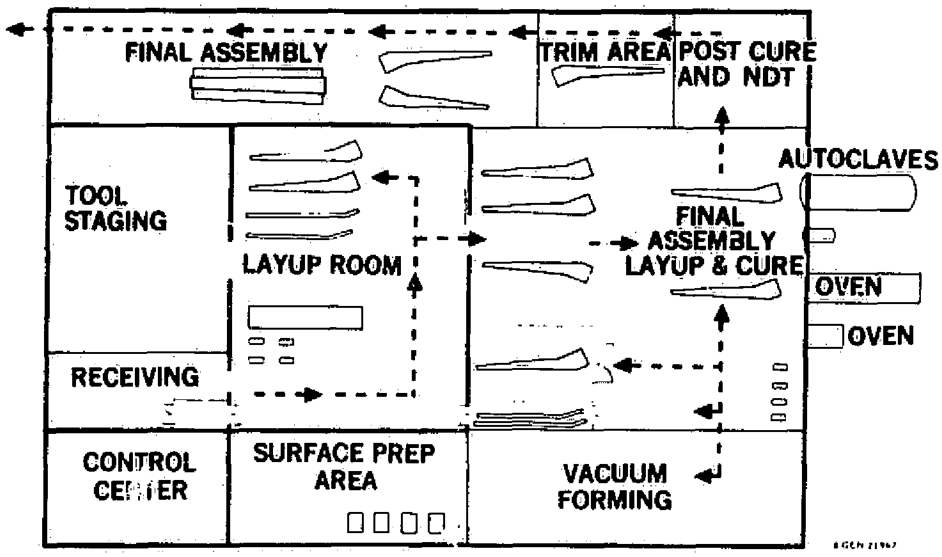


FIGURE 7-13. COMPOSITE WING DEVELOPMENT FACILITY

by fasteners in a manner similar to normal production methods.

Fabrication Plans – Specially fabricated metallic details will be required in three areas: titanium landing gear fitting cocured to the wing skins (upper and lower), aluminum bulkheads attaching upper and lower titanium landing gear webs, and aluminum splice plates at the wing centerline area.

Titanium doublers are designed to straight-taper in three directions and may therefore be machined from 18-centimeter (7-inch) plate stock and, by design, will require numerically controlled fabrication. Use of forgings for initial stock requirement is being considered. Splice plates will be straight-tapered stock, with no unusual machine work anticipated.

Of the 22 ribs, 21 will be composite layups with edge-reinforced lightening holes for access during assembly. Attachment to intercostals will be by bolt-type fasteners.

Leading and trailing edge spars will be basically flat layups (with aerodynamic break) stiffened vertically by molded, blade-type stiffeners in parallel rows. Spars will run uninterrupted from the centerline of the aircraft to the wing tip.

Along the front spar at each location, a composite cup-shaped part will be bonded for clearance of the slat guide fitting in the fully retracted position.

The cup will afford fully encapsulated coverage of the aft end of the slat mechanism.

Localized material buildups will be incorporated during the rear spar layup for flaps, spoilers, and hinges. The design of the spar and buildups will accommodate existing components.

The wing upper surface will contain openings for six access holes. Three outboard holes, oval in shape, will be interchangeable, and three inboard holes of larger size will be interchangeable. On the lower surface, 10 access holes in the midwing area will also be interchangeable with each other.

Numerous fuel probe locations will require small, configured access covers.

Fabrication Outline — Each integrally cured section will be fabricated individually on special tooling with provision for contours. The curvature of the upper and lower skins differs; additionally, a controlled amount of twist is designed into the wing to compensate for torsional deflection of the wing from flight loads.

The width of the wing box varies from 2.75 meters (9 feet) at the fuselage to 0.9 meter (3 feet) at the tip. Broad goods currently available in widths of 122 or 152 centimeters (48 or 60 inches) will be purchased preimpregnated with a B-staged resin system (5208). The material will be spooled so that a full semispan length [18 meters (60 feet)] can be placed upon a skin tool in one continuous ply. Preplied graphite/epoxy is presently available from companies such as Hercules in $\pm$45-degree, 0-degree, and 90-degree orientations. Where convenient, the ply orientation for the skin will be preplied by the supplier to facilitate fast, easy layup on a tool.

By pulling the material lengthwise along the tool, only a simple cutoff operation is required to rapidly build up section thickness. The use of preplied materials precludes the use of many individual Mylar templates which is a time-consuming process both for layup and for verification by inspection.

Once the skin has been applied to the tool, mandrels wrapped with channel shaped graphite sections will be placed on the skin. Between each channel, longitudinal 0-degree preplied strips will be placed to provide the bending resistance for the blades. Each of these layups will be built up to form the blades over the skin (Figure 7-14). The channels will be stacked up from one edge of the skin to the other edge. A total of 12 blade stiffeners per skin was included in our conceptual design.

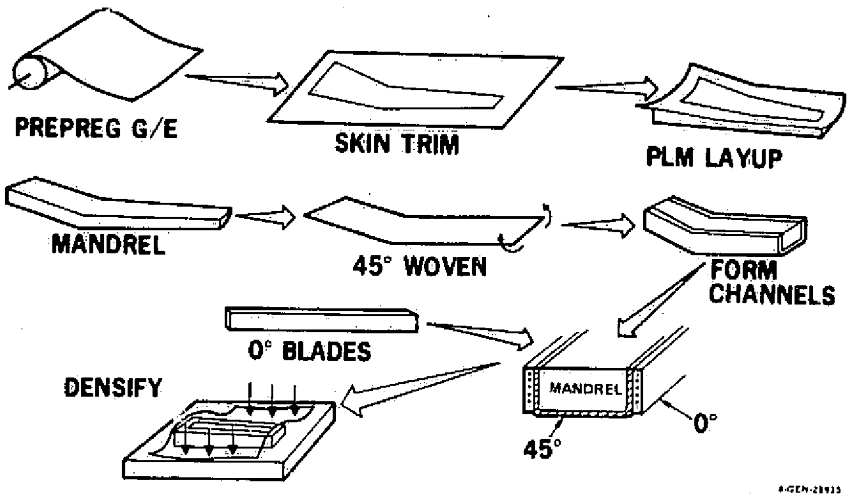


FIGURE 7-14. WING SKIN LAYUP

Between each blade stiffener, running fore and aft, are 22 intercostals that act as tie-in points rib clips. The intercostals will be back-to-back angles in cross section that cure as inverted tees (Figure 7-15). Aluminum blocks faced with silicone rubber approximately 1.25 cm (1/2 inch) thick will serve as molding forms for the intercostals. The intercostals will be applied to the ends of the blocks as an angle in cross section. Each block will be placed between the blade channels and the block will be aligned and anchored so as to maintain dimensional accuracy and precise location of the graphite components during cure. Either longitudinal metal members will align the blocks or a cover plate with alignment lugs will correctly position each block (Figure 7-16). As heat for the curing cycle elevates the rubber facing temperature, expansion of the rubber will provide horizontal pressure that molds the blades and intercostal simultaneously. Pressure to cure the skin is derived from the autoclave.

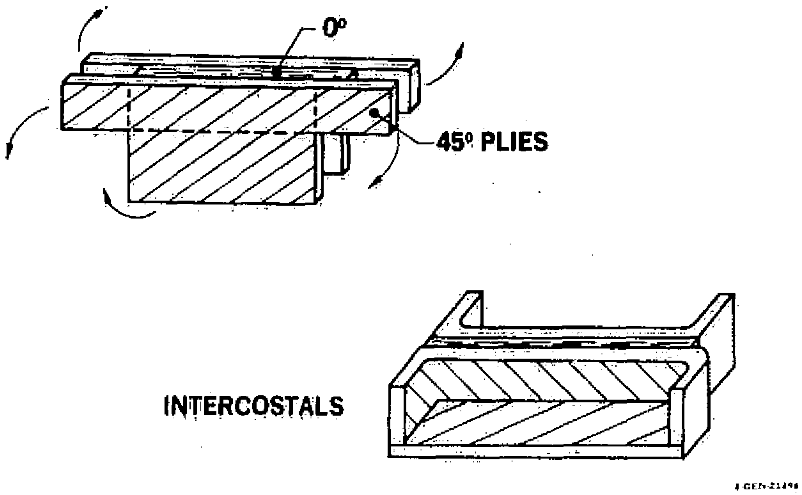


FIGURE 7-15. COVER SKIN INTERCOSTAL CONSTRUCTION

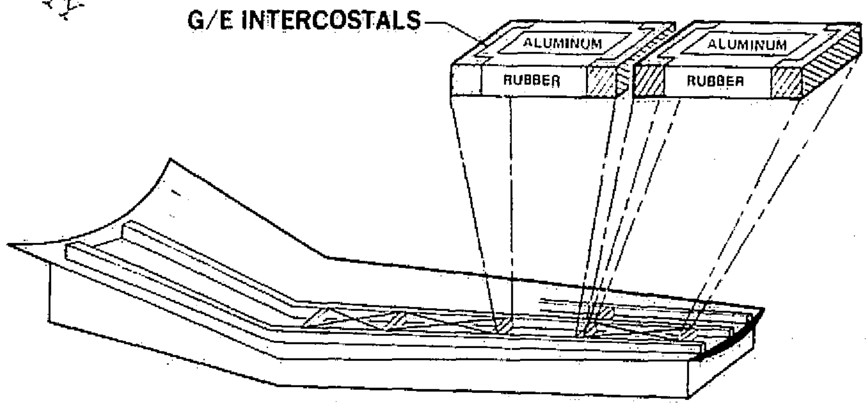


FIGURE 7-16. SKIN-STIFFENER LAYUP

After all the blocks are locked in position, the complete tool and graphite layup will be vacuum bagged with a silicone rubber blanket and cured in the autoclave at 0.69 MPa (100 psig). The time-temperature cycle will be established by Materials and Process Engineering. Thermocouples imbedded in the thermal lagging areas of the part will be monitored on a permanent record to document the cure cycle.

During the manufacturing and tooling development, it may be indicated that supplementary heaters are necessary to improve the heating rates of the blade mandrels. Such heaters could readily be incorporated at multiple locations in the metal mandrels and wired to automatic microprocessor controls similar to the method on the DC-10 upper aft rudder program.

The conceptual design employed separate chordwise ribs of varying cross sections with lightening holes and integral stiffeners. Because of the generally flat shape of the ribs, a heated press would be a logical method for curing these parts.

Part layup could be accomplished with automated equipment, with the localized hole stiffeners manually added to the laminate. Integral stiffeners cured with the ribs can be molded by using silicone rubber pads faced with aluminum at the stiffener contact face. Horizontal pressure is produced by thermal expansion of the rubber reacting against the molding tool boundaries. Figure 7-17 presents a schematic of a simplified molding arrangement.

The front and rear spars are flat, shear-resistant webs with vertical stiffening elements added after cure. The cross-sectional thickness of the spars changes with length, which lends itself to simple layup on a flat tool surface. Buildups for attachment points will be added to locally increase the spar thickness. An autoclave will be used to cure the spars.

Manufacturing will receive prepreg material (tape and broad goods) ready for use. Prepreg rolls are stored at $-18^{\circ}C$ ($0^{\circ}F$) until required for actual layup operations.

Table 7-6 outlines the fabrication sequence for the wing elements.

Mylar templates can be used with waterjet cutting equipment to rapidly trim plies to required profiles. Multiple stacks of graphite/epoxy prepreg have been cut with a water jet at Douglas. The effect of water on the prepreg has been evaluated by Materials and Process Engineering, and it is not considered detrimental to part quality.

Material plies will be applied to the plastic laminating mold (PLM) with verification of each ply position in the stackup controlled by the fabrication orders and Quality Assurance inspectors. Skins laid out on the PLM will then be put on the titanium landing gear doubler and covered (after the final ply) with mandrels for location blade stiffeners and intercostals.

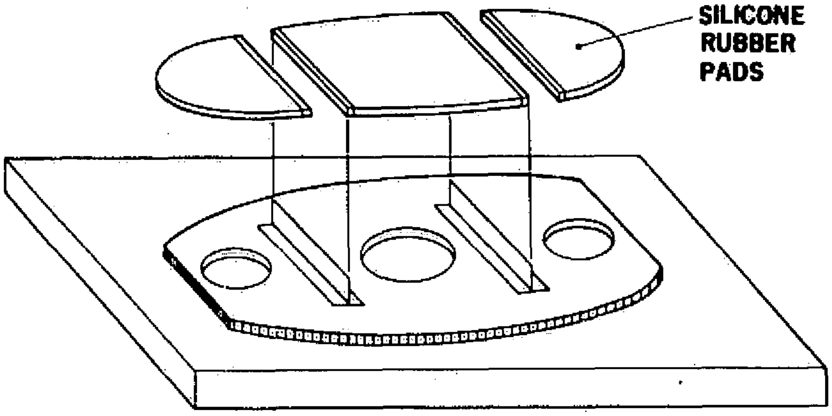


FIGURE 7-17. MOLDING RIBS

TABLE 7-6
FABRICATION SEQUENCE

	SKIN	BLADE STRINGERS	INTERCOSTAL	RIBS
MATERIAL:	BROADGOODS	BROADGOODS PLUS TAPE	BROADGOODS	BROADGOODS
TOOLS	SKIN MOLD 	PREFORM MANDRELS 	BLOCK MANDREL 	MOLD 
FABRICATION	LAYUP METAL INSERTS DENSIFY	45° CHANNELS 0° BLADES DENSIFY	LAYUP DENSIFY	LAYUP DENSIFY

The PLM will be kept in a -18°C (0°F) freezer when not being actively worked upon, thereby retaining the workability of the prepreg material. On completion of layup, the part and tool will be vacuum bagged and cured in the autoclave under heat and positive pressure.

Access doors will be fabricated by hand layup and trimmed by router to portable trim tool dimensions.

Slat track cups will be laid up in an exterior mold with a silicone rubber plug and cured in small quantities with the larger parts.

Spars will be long, narrow, flat layups applied to a PLM with the required aerodynamic break. Mandrels will control stiffener shape and access hole definition. Both forward and aft spars will be cured on the same substructure on side-by-side PLM tools.

Immediately following cure, composite parts will be trimmed by diamond saw, track-mounted router, or tracer router tooling. Holes will be drilled to size utilizing drill jigs and portable drilling equipment.

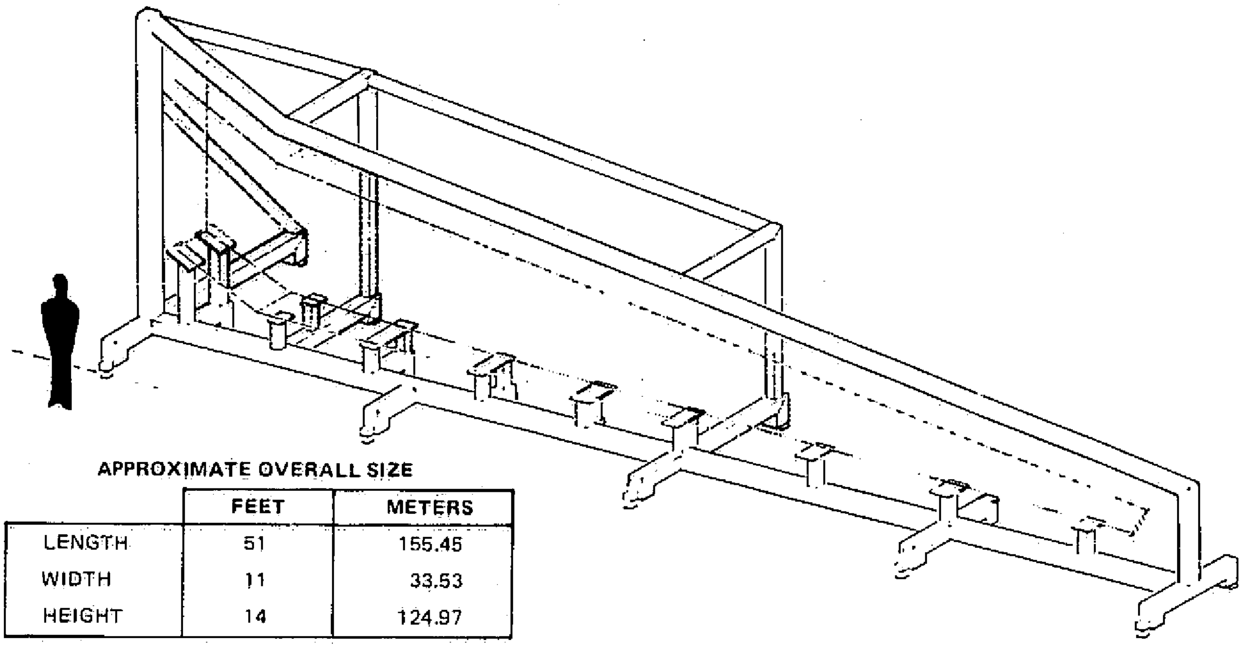
During the final fabrication and assembly operations, coupons will be provided from trimmed excess material for evaluation by Quality Assurance.

Assembly Outline — Final assembly will be accomplished in a vertical jig with the wing oriented forward spar down, as shown in Figure 7-18. The upper skin will be loaded first with ribs, the spars following. The final step will be the application of the lower skin panel, slat track cups, centerline splice webs, and access panel cover.

The completed box will be subjected to inspection and nondestructive evaluation. Upon acceptance as a structurally sound unit, the remaining assembly operations will be performed, including leading and trailing edge attachment; installation of hinge fittings, slats, flaps, spoilers, cable runs, and fuel components; and final painting.

A flow diagram of the assembly sequence is shown in Figure 7-19.

All operations will be controlled by assembly outline documentation and tracked on 125A position control charts by Industrial Engineering personnel.



APPROXIMATE OVERALL SIZE

	FEET	METERS
LENGTH	51	155.45
WIDTH	11	33.53
HEIGHT	14	124.97

FIGURE 7-18. WING ASSEMBLY JIG

8-GEN-21908

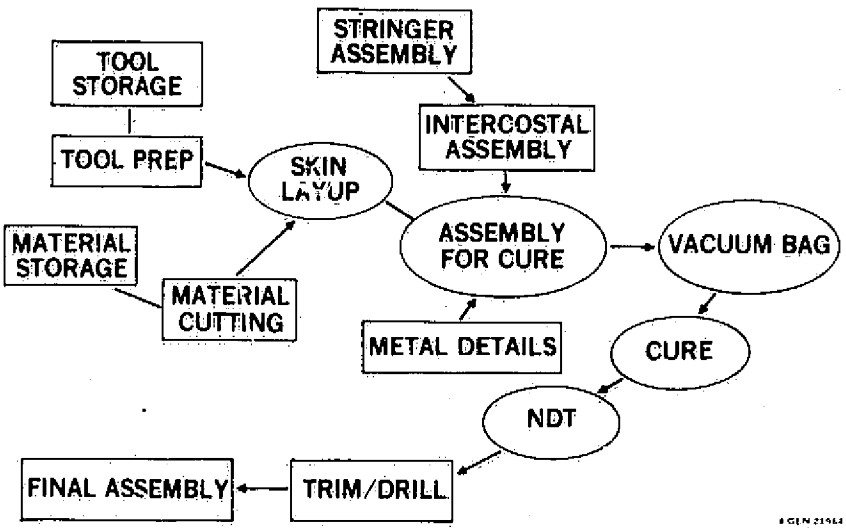


FIGURE 7-19. ASSEMBLY FLOW

Attachments through graphite/epoxy will be made through clearance holes with titanium fasteners installed wet. The sealant acts to prevent fuel leakage and to minimize galvanic corrosion between the graphite and fastener.

On completion of the cure cycle, the assembly will be transferred to the trim area for trimming of edges and mating surfaces by a diamond saw. Holes for fastener installations will be located by a jig and drilled following trim operations.

Holes have been effectively drilled through graphite/epoxy using "dagger" drills of solid carbide at 2000 rpm. These drills resemble spade drills with special modifications of the cutting edge and a sharp included point angle. One of the advantages of this style of drill is the minimum breakout on the back of the graphite/epoxy and the elimination of reaming because of close tolerance and good surface finish as drilled. By avoiding the reaming operation, both tooling and time are saved.

Composites may be drilled and trimmed dry; however, the tool life is enhanced by using liquid or spray coolants. The dust problem associated with composites is reduced by the application of coolants. Where coolants cause contamination for secondary bonding, vacuum pickups can be used at the work site.

Tooling Plans — Skin and parts layup will be placed on a contoured plastic laminating mold for curing operations. The contour will simulate lofted surfaces of the parts during cure, and will be mounted on an egg-crate type structure for rigidity. The surface contour of the skin PLM and spar PLM tools will be derived by rolling or braking sheet aluminum to rough contour, and final precision finishing surfaces by N/C milling to the required shape and contour.

Smaller PLM tools will allow fabrication of access doors and small parts suitable for curing in available autoclaves other than the large, wing-sized unit.

Aluminum machined mandrels covered with a uniform coating of silicone rubber will be fabricated for molding interstices between stiffeners, intercostals, spar buildups, and access panel openings. Mandrels will be N/C machined to contour with programmed allowances for coefficient of expansion and rubber encapsulation thickness. In areas with mandrels, cal-rod heaters will assist in maintaining uniform distribution and proper temperature rise.

Mylar drawings produced by the Gerber plotter will provide direct full-size patterns for prepreg layup, positioning, inspection, and placement on the PLM. The same N/C data which produced the numerical control draft tools will also direct the water-jet trimming machine. Drafts will be produced through the Computer-Aided Design Tooling (CADT) process to keep layout time at a minimum cost.

Nondesigned tooling will be provided for drilling fastener locations, trimming access openings, panel edges, slat hinge cups, and splice plate details. Normally, these will be flat sheet metal tools or fiberglass blanket-type tools with bushed hole locations and steel trim bar edges referenced from master tooling shapes and flat master layouts.

Hard master tooling for control of hinge line, contour source, and hinge location will utilize existing master tooling now provided for the conventional wing.

The wing will be held during final assembly utilizing a vertical (leading edge down) assembly jig constructed to permit loading and fastening of bulkheads, spars, and fasteners in one position. Movement of composite layups and parts within the fabrication areas will be accomplished by the use of a variety of rolling table and rack-type fixtures, designated as handling fixtures.

Overhead rail-mounted cranes will move and position jigs, bulk material rolls, tools, parts, and assembled wing boxes.

Phase V – Composite Wing Box Flight Preparation

To prepare the composite wing for flight evaluation, a DC-9 Series 32 aircraft will be obtained and modified for installation of a composite wing box. The existing left-hand wing, from the center wing splice outboard, will be removed from the aircraft. The aluminum right-hand wing will remain in place with splices incorporated at the centerline junction of both halves. As this is a wet wing, appropriate fuel sealing methods will be used to prevent leakage. Plumbing for fuel supply must be added to the composite wing box and integrated into the existing onboard systems. The left-hand control surfaces, fairings, and leading and trailing edges from the existing metal wing will be used. Similarly, the hydraulic systems and landing gear are normal production items that will be installed by production personnel who have become familiar with graphite/epoxy materials.

The manufacture of the left-hand composite wing box for flight development is charged to Phase V. The flight composite wing box will be manufactured in a similar manner as the three components produced in Phase III for laboratory test.

Some engineering modification drawings for metal and composite parts are anticipated to facilitate installation of the composite wing box on an existing DC-9-32. These special parts will be fabricated in accordance with existing procedures for metal parts and according to Phase III composite procedures for the composite parts. Since Phase III tasks will have been already accomplished, the production of any special composite parts should provide no particular problems.

TEST PLAN

Figure 7-20 presents the overall test program and task relationships from design requirements for tests through aircraft FAA flight certification.

Some technical development for test purposes is anticipated for this program as a result of the use of composite materials and composite design and production techniques. Attachment of load application fittings to an aerodynamic surface formed of composite material is an example of an item to be tested in technical development. Load application to conventional metal aerodynamic components (wings, stabilizers, etc.) is normally accomplished by attaching load fittings to the structure for hydraulic jacks by removing normal production fasteners — rivets, screws, and bolts — and attaching the fittings using fasteners in the vacated holes. Composite aerodynamic surfaces, however, will have a greatly reduced number of fasteners which can be removed and used for load fitting attachments. Accordingly, it is anticipated that a certain amount of test design and development will be required to produce acceptable fitting attachment methods which do not adversely or unrealistically affect the specimen strength or fatigue life.

C-3

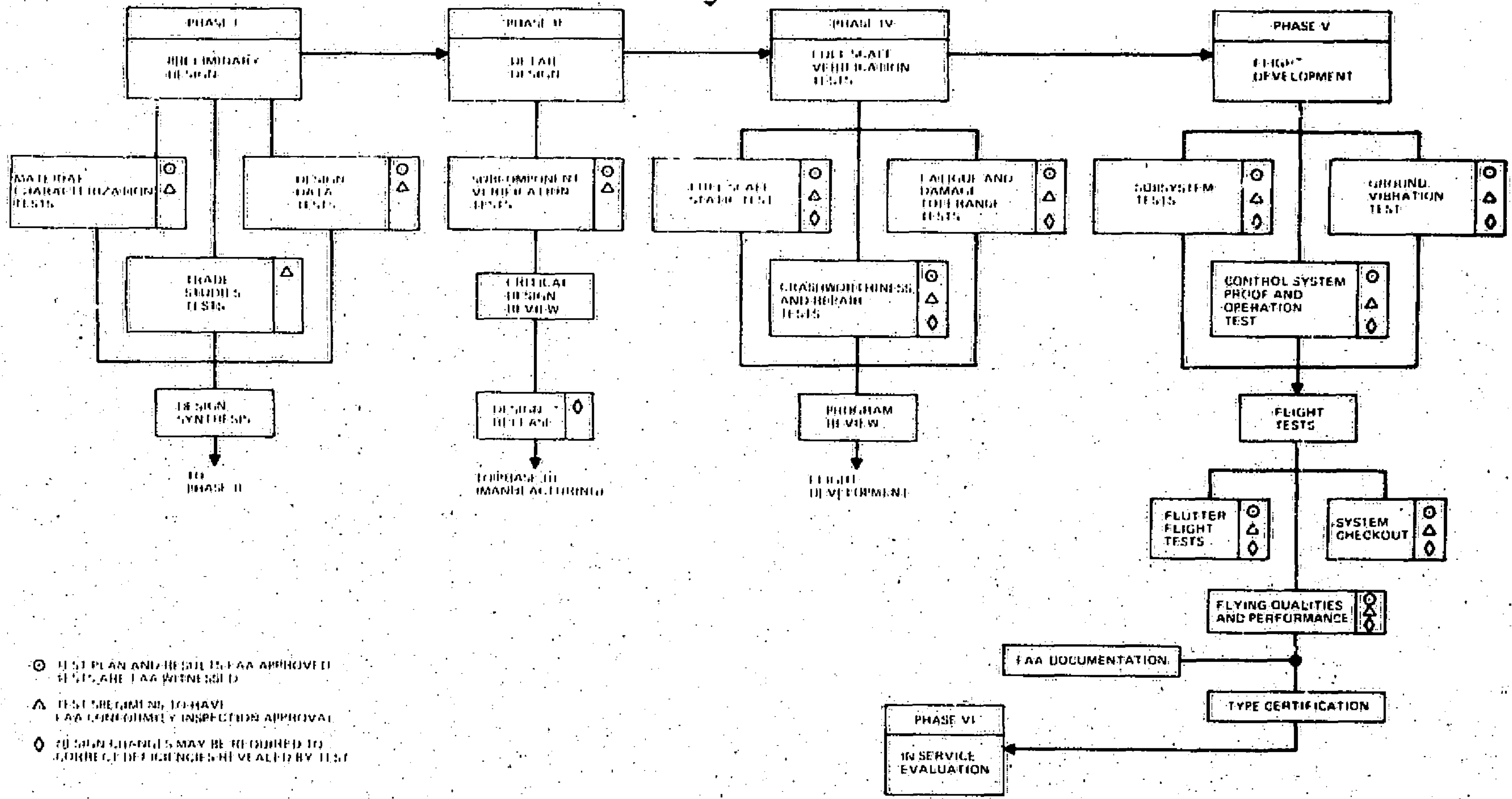


FIGURE 7-20. TEST FLOW DIAGRAM

Material allowables and design verification tests will be performed to demonstrate compliance with applicable requirements of Federal Aviation Regulations, Part 25, and the current FAA Composite Structures Guidelines. FAA will inspect the test articles for design conformity, approve the test plans including load conditions, witness the test, and approve the final test report.

Final reports of the test results and, as appropriate, their correlation with the predicted values will be prepared.

Design Development Tests

A design development program will be conducted in Phase I to determine composite material properties and structural component performance that are not available in handbooks or other approved sources and to develop design concepts that will meet strength, damage tolerance and fatigue, lightning strike, and static electricity requirements.

The development test program will be determined considering the background available for composite material from research, in-house composite programs, other programs in industry, and government agencies. A representative structural development test program is presented in the following paragraphs.

Structural Design Development Tests — The structural design development plan includes concept development testing of critical structural elements, joints, and fittings, and testing to determine laminate mechanical properties and fracture mechanics data. Preliminary design studies will lead to the definition of several candidate design elements, joints, and fittings. These candidates will undergo development testing to determine comparative performances of different concepts.

Critical structural elements of the composite wing box are to be selected for design development testing. Tables 7-7 through 7-10 illustrate typical test specimens and conditions for concept evaluation for wing skin panels, spar and rib webs, joints, and fittings. The 260 specimens illustrated with 25 different types of design detail sections are considered representative of a concept design development program for a DC-9-32 composite wing box. More than one configuration, as noted in the column entitled "Structural Concept" in Tables 7-7 through 7-10, would be tested for a given detail section. Typical differences in configuration might be, for example, differences in element dimensions, number and orientation of plies, or stiffener depth.

TABLE 7-7

SKIN PANEL CONCEPT DESIGN DEVELOPMENT TESTS

TEST NO.	TEST SPECIMENS	SPEC SIZE cm x cm (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP °C (°F)	PRETEST CONDITIONING PERCENT RH	NO. OF SPECIMENS TEST TEMP °C (°F) −54 (−65)	NO. OF SPECIMENS TEST TEMP °C (°F) AMB	NO. OF SPECIMENS TEST TEMP °C (°F) 82 (180)	NO. OF SPECIMENS TOTAL
1	BASIC PANELS	122 x 61 (48 x 24)	TENSION STRENGTH AND STIFFNESS	LONGITUDINAL TENSION	6	82 (180) AMB	95 AMB	2	7	2	11
2		122 x 229 (48 x 90)	COMPRESSION STRENGTH AND STIFFNESS	LONGITUDINAL COMPRESSION	6	82 (180) AMB	95 AMB	2	7	2	11
3			SHEAR STRENGTH AND STIFFNESS	IN-PLANE SHEAR	6	82 (180) AMB	95 AMB	2	7	2	11
4		122 x 229 (48 x 90)	BENDING STRENGTH	NORMAL PRESSURE	6	AMB	AMB		6		6
5			STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	3	AMB	AMB		9		9
6			STRENGTH UNDER COMBINED LOADING	COMPRESSION AND SHEAR	3	AMB	AMB		9		9
7			STRENGTH UNDER COMBINED LOADING	COMPRESSION AND NORMAL PRESSURE	3	AMB	AMB		3		3
8		36 x 61 (14 x 24)	FATIGUE STRENGTH	LONGITUDINAL R = −1.0	3	AMB	AMB		3		3
9	ACCESS PANEL	122 x 229 (48 x 90)	TENSILE STRENGTH	LONGITUDINAL TENSION	2	AMB	AMB		2		2
10			COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	AMB	AMB		2		2
11			SHEAR STRENGTH	IN-PLANE SHEAR	2	AMB	AMB		2		2
12			COMBINED COMPRESSION AND SHEAR	LONGITUDINAL COMPRESSION AND IN-PLANE SHEAR	2	AMB	AMB		6		6
13			FATIGUE STRENGTH	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
14			EFFECTS OF FUEL PRESSURE	NORMAL PRESSURE	2	AMB	AMB		2		2
15	DAMAGED PANEL (SMALL AREA)	36 x 61 (14 x 24)	POSTDAMAGE TENSION STRENGTH	LONGITUDINAL TENSION	2	82 (180) AMB	95 AMB	2	3	2	7
16		122 x 229 (48 x 90)	POSTDAMAGE COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	82 (180) AMB	95 AMB	2	3	2	7
17		36 x 61 (14 x 24)	POSTDAMAGE FATIGUE	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
18	DAMAGED PANEL (LARGE AREA)	36 x 61 (14 x 24)	POSTDAMAGE TENSION STRENGTH	LONGITUDINAL TENSION	2	82 (180) AMB	95 AMB	2	3	2	7
19		122 x 229 (48 x 90)	POSTDAMAGE COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	1	82 (180) AMB	95 AMB	[illegible]	3	2	7
20		36 x 61 (14 x 24)	POSTDAMAGE FATIGUE	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
21	REPAIRED PANEL (SMALL AREA)	36 x 61 (14 x 24)	TENSILE STRENGTH	LONGITUDINAL TENSION	2	82 (180) AMB	95 AMB	2	3	2	7
22		122 x 229 (48 x 90)	COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	82 (180) AMB	95 AMB	2	3	2	7
23		36 x 61 (14 x 24)	FATIGUE STRENGTH	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
24	REPAIRED PANEL (LARGE AREA)	36 x 61 (14 x 24)	TENSILE STRENGTH	LONGITUDINAL TENSION	2	82 (180) AMB	95 AMB	2	3	2	7
25		122 x 229 (48 x 90)	COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	2	82 (180) AMB	95 AMB	2	3	2	7
26		36 x 61 (14 x 24)	FATIGUE STRENGTH	LONGITUDINAL R = −1.0	2	AMB	AMB		2		2
27	SWEEP BREAK PANEL	61 x 152 (24 x 60)	TENSION STRENGTH	LONGITUDINAL TENSION	3	AMB	AMB		3		3
28			COMPRESSION STRENGTH	LONGITUDINAL COMPRESSION	3	AMB	AMB		3		3

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TABLE 7-8

SPAR AND RIB CONCEPT DESIGN DEVELOPMENT TEST

TEST NO.	TEST SPECIMENS	SPEC SIZE cm x cm (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP °C (°F)	PRETEST CONDITIONING PERCENT RH	NO. OF SPECIMENS TEST TEMP °C (°F) −54 (−65)	AMB	82 (180)	TOTAL
1	SPAR WEB 	61 x 183 (24 x 72)	WEB SHEAR STRENGTH AND STIFFNESS	3-POINT BEAM BENDING	6	82 (180) AMB	95 AMB	2	7	2	11
2			WEB LATERAL BENDING STRENGTH	LATERAL PRESSURE	3	AMB	AMB		3		3
3	RIB AND BULKHEADS 		WEB SHEAR STRENGTH AND STIFFNESS	IN-PLANE SHEAR	6	82 (180) AMB	95 AMB	2	7	2	11
4			WEB LATERAL BENDING STRENGTH	LATERAL PRESSURE	3	AMB	AMB		3		3

TABLE 7-9

JOINT CONCEPT DESIGN DEVELOPMENT TEST

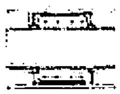
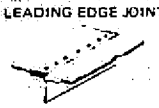
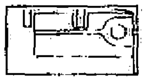
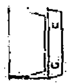
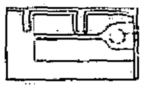
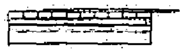
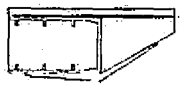
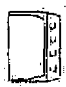
TEST NO.	TEST SPECIMENS	SPEC SIZE cm x cm (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP °C (°F)	PRETEST CONDITIONING PERCENT RH	NO. OF SPECIMENS TEST TEMP °C (°F) −54 (−65)	AMB	82 (180)	TOTAL
1	WING TO FUS JOINT 	30 x 61 (12 x 24)	SHEAR STRENGTH	STATIC SHEAR	3	82 (180) AMB	95 AMB	2	4	2	8
2	CENTERLINE JOINT 	51 x 76 (20 x 30)	TENSION STRENGTH	LONGITUDINAL TENSION	3	82 (180) AMB	95 AMB	2	4	2	8
3			FATIGUE STRENGTH	LONGITUDINAL R = −1.0	3	AMB	AMB		3		3
4	SPAR WEB TO PANEL JOINT 	25 x 76 (10 x 30)	SHEAR STRENGTH	STATIC SHEAR	6	82 (180) AMB	95 AMB	2	7	2	11
5	RIB TO PANEL JOINT 	30 x 61 (12 x 24)	SHEAR STRENGTH	STATIC SHEAR	6	82 (180) AMB	95 AMB	2	7	2	11
6			FUEL PRESSURE RESISTANCE	LATERAL TENSION	1	AMB	AMB		1		1
7	RIB TO SPAR JOINT 		SHEAR STRENGTH	STATIC SHEAR	6	82 (180) AMB	95 AMB	2	7	2	11
8			FUEL PRESSURE RESISTANCE	LONGITUDINAL TENSION	3	AMB	AMB		3		3
9	LEADING EDGE JOINT 	25 x 76 (10 x 30)	STATIC STRENGTH	BIAXIAL TENSION	3	AMB	AMB		3		3
10			EFFECT OF THERMAL MISMATCH	TENSION CYCLING R = 0.1	3	82 (180) AMB	95 AMB	7		(7)	7
11	TRAILING EDGE JOINT 		STATIC STRENGTH	BIAXIAL TENSION	3	AMB	AMB		3		3

TABLE 7-10
FITTING CONCEPT DESIGN DEVELOPMENT TEST

TEST NO.	TEST SPECIMENS	SPEC SIZE cm x cm (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP °C (°F)	PRETEST CONDITIONING % RH	NO. OF SPECIMENS TEST TEMP °C (°F) −54 (−65)	NO. OF SPECIMENS TEST TEMP °C (°F) AMB	NO. OF SPECIMENS TEST TEMP °C (°F) 82 (180)	NO. OF SPECIMENS TOTAL
1	AILERON FITTING 	20 x 51 (8 x 20)	STATIC STRENGTH	MAX COMBINED LOADS	3	AMB	AMB		3		3
2	SPOILER SUPPORT 	30 x 61 (12 x 24)			3	AMB	AMB		3		3
3	FLAP FITTING 	20 x 51 (8 x 20)			3	AMB	AMB		3		3
4	LANDING GEAR FITTING 	25 x 254 (10 x 100)			3	AMB	AMB		3		3
5	TRAP PANEL INSTL 	61 x 102 (24 x 40)			3	AMB	AMB		3		3
6	SLAT TRACK SUPT 	30 x 61 (12 x 24)			3	AMB	AMB		3		3

Design development tests are to be performed to establish laminate configurations and obtain laminate properties data that will meet the strength, fatigue, and damage tolerance requirements for a composite wing box. This is to be done by selecting basic laminate configurations, fabricating coupons representative of the design, and testing these coupons with static, dynamic, and repeated loads to determine laminate properties design data. Approximately 12 test specimen configurations, as shown in Table 7-11, with a total of 1346 specimens are to be tested in the structural design development test program.

Lightning and Static Electricity Tests — A lightning test will be conducted with restrike tests on fastener heads for fuel ignition hazards and lightning transient tests on critical electrical wiring components for transient suppression/shielding designs.

A static electricity test will evaluate the static charge dispersion characteristics of graphite composite fuel tank structures. A static charge spray test or tank will be fabricated. The proposed protection techniques will be evaluated and demonstrated for their effectiveness.

Design Verification Tests

Design verification tests are to be conducted in Phase II on panels, component sections, joints, fittings, and the landing gear attachment to verify that design details from the development tests satisfy the design and FAA requirements. These tests are to be completed before the engineering drawings are released for fabrication of the first full-scale composite wing box for static test.

The results of the design development tests and previous composite component development programs will be utilized to design detail parts for a full-scale composite wing box. Subcomponents representative of the final design will be tested to verify the static strength and fatigue and damage tolerance characteristics of critical design details of the composite wing box and to demonstrate satisfactory repairability of a wing panel including stiffeners. Table 7-12 presents typical design detail verification tests. These tests will be initiated as soon as possible after development tests conducted on a particular design detail are completed.

TABLE 7-11

LAMINATE PROPERTIES DESIGN DEVELOPMENT TESTS

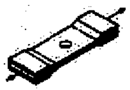
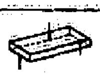
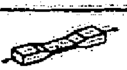
TEST NO.	TEST SPECIMENS	SPEC SIZE cm x cm (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP °C (°F)	PRETEST CONDITIONING PERCENT RH	NO. OF SPECIMENS TEST TEMP °C (°F) −54 (−65)	NO. OF SPECIMENS TEST TEMP °C (°F) AMB	NO. OF SPECIMENS TEST TEMP °C (°F) 82 (180)	TOTAL
1	THICK LAMINATES 	2.5 x 27.9 (1 x 11)	TENSION PROPERTIES	TENSION	10	82 (180) AMB	95 AMB	66	132	66	264
2	THICK LAMINATES 	7.6 x 22.9 (3 x 9)	COMPRESSION PROPERTIES	COMPRESSION	10	82 (180) AMB	95 AMB	66	132	66	264
3	HOLE OUT 	2.5 x 27.9 (1 x 11) HOLE 0.476 (3/16)	TENSION ALLOWABLE	TENSION	3	82 (180) AMB	95 AMB	24	48	24	96
		6.4 x 27.9 (2.5 x 11) HOLE 1.27 (1/2)			3	82 (180) AMB	95 AMB	24	48	24	96
		12.7 x 38.1 (5 x 15) HOLE 2.5 (1.0)			3	82 (180) AMB	95 AMB	24	48	24	96
4		7.6 x 27.9 (3 x 11)	JOINT STRENGTH VARIABLES • 3 FASTENER SIZES • 3 THICKNESSES • 2 W/D STAGGERED PATTERN	STATIC TENSION	3	82 (180) AMB	95 AMB	12	57	12	81
				FATIGUE LIFE R = ±0.5, −1	3	AMB	AMB		108		108
5			BEARING AND SHEAR-OUT	TENSION	3	AMB	AMB	9	18	9	36
6		TBD	FRACTURE TOUGHNESS	BENDING	3	82 (180) AMB	95 AMB	10	20	10	40
7		TBD	FRACTURE TOUGHNESS	TENSION	3	AMB	AMB	5	15	5	25
8		1.5 x 0.6	INTERLAMINAL SHEAR	BENDING	3	82 (180) AMB	95 AMB	5	20	5	30
9		6.4 x 27.9 (2.5 x 11)	FATIGUE • WITHOUT HOLE • WITH HOLE R = ±0.5, −1	TENSION AND COMPRESSION	3	82 (180) AMB	95 AMB	20	60	20	100
10		7.6 x 22.9 (3 x 9)	EFFECT OF IMPACT (2 LEVELS) 16.9 N · M (150 IN.-LB) 14.7 N · M (130 IN.-LB)	COMPRESSION	2	82 (180) AMB	95 AMB	10	30	10	50
11		7.6 x 27.9	EFFECT OF IMPACT (2 LEVELS)	TENSION	2	AMB	AMB		20		20
12		7.6 x 27.9	FRACTURE TOUGHNESS	TENSION	3	82 (180) AMB	95 AMB		40		40

TABLE 7-12

DESIGN VERIFICATION TESTS

TEST NO.	TEST SPECIMENS	SPEC SIZE cm x cm (IN. x IN.)	TEST PURPOSE	TEST LOADING	STRUCT CONCEPT	PRETEST CONDITIONING TEMP °C (°F)	PRETEST CONDITIONING PERCENT RH	NO. OF SPECIMENS TEST TEMP °C (°F) –54 (–65)	AMB	82 (180)	TOTAL
1	WING COVER PANEL 	122 x 152 (48 x 60)	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	82 (180)	95		1		2
				COMPRESSION AND SHEAR	1	82 (180)	95		2		2
				COMPRESSION AND NORMAL PRESSURE	1	82 (180)	95		1		1
2			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	82 (180)	95		2		2
3	WING ACCESS DOOR PANEL 	122 x 152 (48 x 60)	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	82 (180)	95		1		1
				COMPRESSION AND SHEAR	1	82 (180)	95		1		1
4			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	82 (180)	95		1		1
5	DAMAGED PANEL 	122 x 152 (48 x 60)	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	82 (180)	95		1		1
				COMPRESSION AND SHEAR	1	82 (180)	95		1		1
			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	82 (180)	95		1		1
6	REPAIRED WING PANEL 	122 x 152 (48 x 60)	STRENGTH UNDER COMBINED LOADING	TENSION AND SHEAR	1	82 (180)	95		1		1
				COMPRESSION AND SHEAR	1	82 (180)	95		1		1
			FATIGUE UNDER COMBINED LOADING	AXIAL AND SHEAR	1	82 (180)	95		1		1
7	REAR SPAR 	61 x 183 (24 x 72)	WEB STRENGTH	3-POINT BEAM BENDING	1	AMB	AMB		2		2
8	FRONT SPAR 	61 x 183 (24 x 72)	WEB STRENGTH	3-POINT BEAM BENDING	1	AMB	AMB		2		2
9	CENTERLINE JOINT 	51 x 76 (20 x 30)	TENSION STRENGTH	LONGITUDINAL TENSION	1	82 (180)	95		1		1
10			FATIGUE STRENGTH	LONGITUDINAL R = –1.0	1	82 (180)	95		1		1
11	SPAR CAP TO PANEL JOINT 	25 x 76 (10 x 30)	SHEAR STRENGTH	STATIC SHEAR	1	82 (180)	95		2		2
12			FATIGUE STRENGTH	SHEAR R = –1.0	1	82 (180)	95		2		2
13	FLAP FITTING 	20 x 51 (8 x 20)	STATIC STRENGTH	DESIGN ULTIMATE LOAD	1	AMB	AMB		1		1
			FATIGUE STRENGTH	TBD	1	AMB	AMB		1		1
14	TRAP PANEL INSTL 	61 x 102 (8 x 20)	STATIC STRENGTH	DESIGN ULTIMATE LOAD	1	AMB	AMB		1		1
			FATIGUE STRENGTH	TBD	1	AMB	AMB		1		1

Full-Scale Verification Tests

Verification tests are to be performed on two full-scale semispan composite wing boxes and one major subcomponent in Phase IV. The first semispan box will be used to verify design static strength requirements. The major subcomponent article will be used for crashworthiness tests. The second semispan box will demonstrate fatigue life and damage tolerance. The two semispan boxes and the major subcomponent article, after failure or with flaws induced, are to be used to demonstrate repair procedures and strength after repair. Vibration tests are to be performed on all three semispan specimens to determine the composite wing box free/free and installed vibration characteristics.

Full-Scale Wing Static Strength Design Verification Test — A full-scale composite wing box semispan is to be tested to verify the wing box design stiffness design limit strength, and design ultimate strength to DC-9 design specifications for the critical load conditions. The test article will then be loaded to failure for the most critical condition.

The test article will be a structurally complete semispan composite wing box produced by Manufacturing. It will include all structurally significant fittings and access panels and any additional fittings required for handling and test loading. The composite wing box will be joined with a production DC-9 metallic right wing and a DC-9 fuselage center section. Dummy landing gear will be fabricated, installed, and utilized as part of the loading fixture. Inspections will be performed on the composite wing box during its manufacture, assembly, and test setup in accordance with FAA conformity inspection procedures. A typical test setup is shown in Figure 7-21.

Instrumentation will consist of deflectometers, strain gages, load cells, pressure transducers, and associated signal conditioning, calibration equipment, power supplies, cabling, oscilloscopes, and other instruments.

One of the major design goals for this test is to design a composite wing box with a stiffness equivalent to the DC-9 metal wing. Accordingly, deflection data will be obtained on the first two composite wing boxes. Data will be compared to analytical finite model deflection data to verify that deflection characteristics of the composite wing box conform to design requirements.

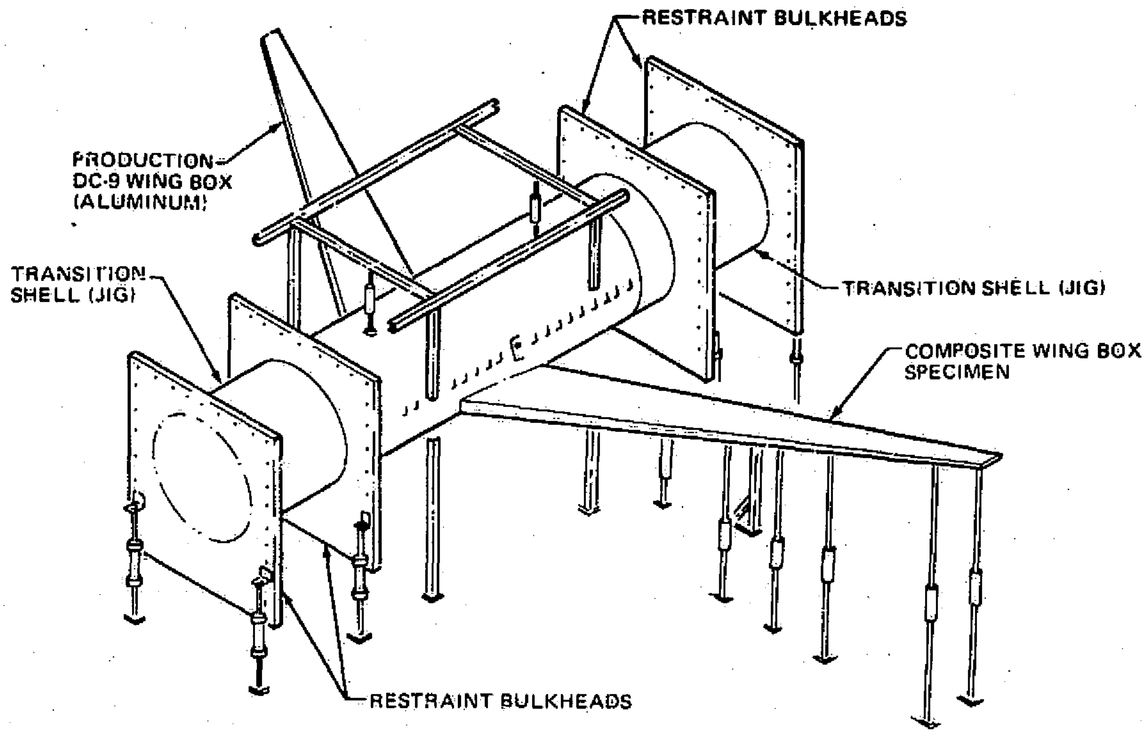


FIGURE 7-21. COMPOSITE WING BOX DESIGN VERIFICATION STATIC TEST

The ultimate strength (failure) test may require load application in excess of the ultimate strength of the balancing metal wing. Accordingly, analyses will be performed to define unsymmetrical bending moments to unload the metal wing and load the composite wing box at the critical locations to produce failure in the composite wing box.

The sequencing for this test is shown in Figure 7-22.

The test plan will be approved and the test witnessed by the FAA for compliance with FAR requirements.

Crashworthiness Design Verification Test — A test will be conducted to demonstrate that landing gear failure due to overloads during takeoff or after landing (assuming overloads act in the upward and aft direction) does not result in fuel

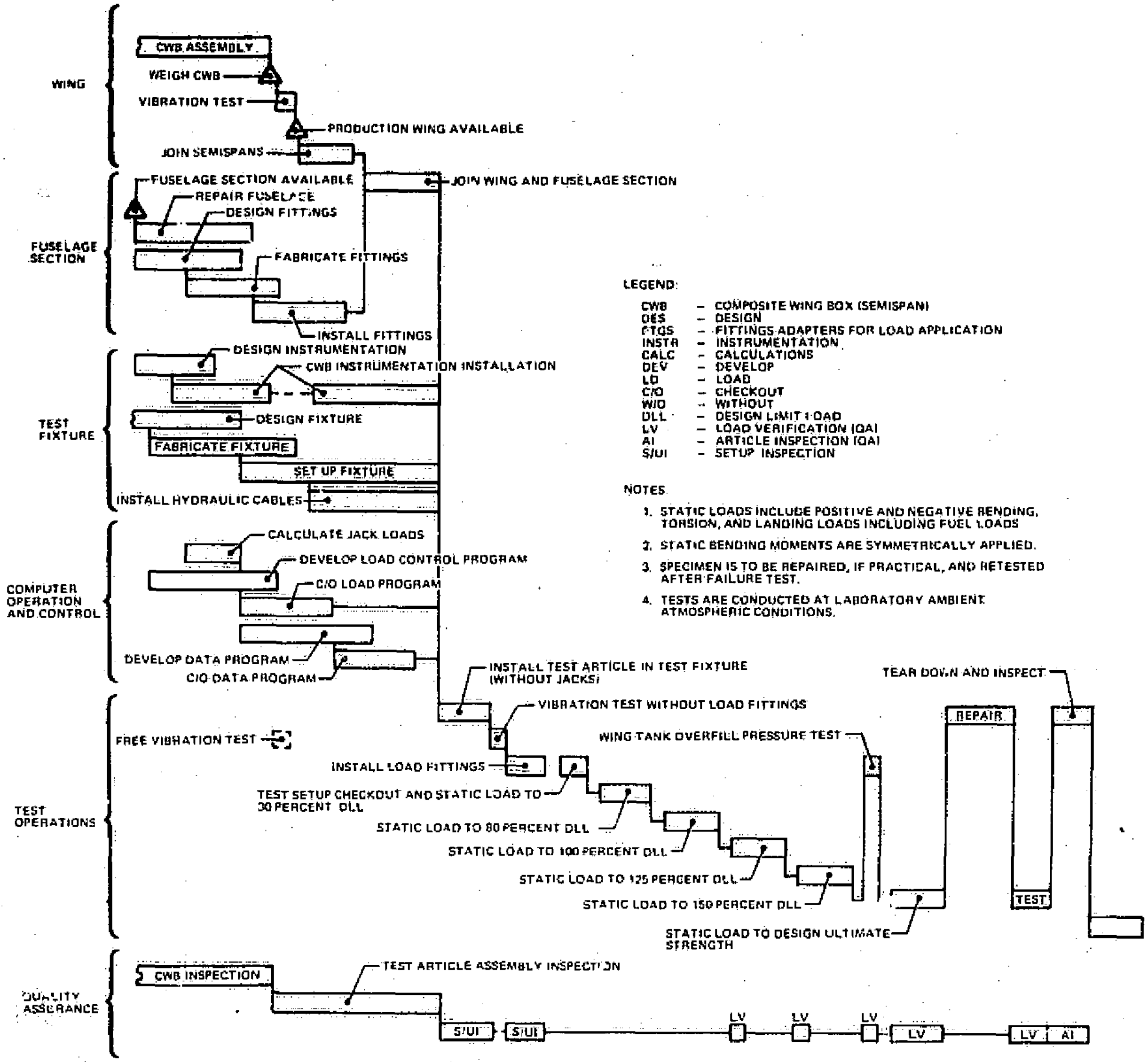


FIGURE 7-22. COMPOSITE WING STATIC TEST PROGRAM

spillage from the wing tank sufficient to constitute a fire hazard. [See Para. 25.721 (a) (2) of Reference 3.]

Verification that the main landing gear will separate from the wing without significant fuel spillage will be demonstrated by test on a typical composite wing box section, as shown in Figure 7-23, with a dummy landing gear and landing gear side brace. The test plan will be approved and the test witnessed by the FAA for compliance with Federal Aviation Regulations.

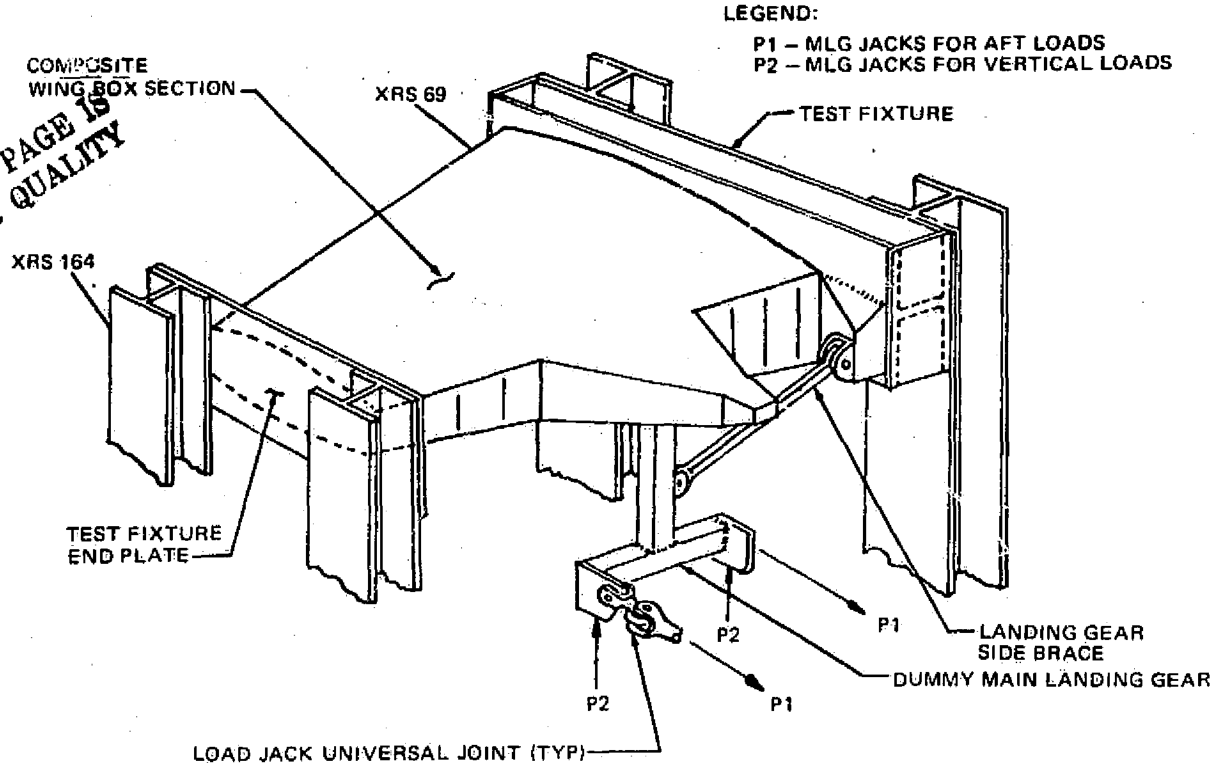


FIGURE 7-23. LANDING GEAR CRASHWORTHINESS TEST SETUP

Repairs will be made of damage resulting from the intital crashworthiness tests, and the specimen will be tested again to determine the static strength of the repairs.

Full-Scale Wing Fatigue and Damage Tolerance Design Verification Tests – Tests are to be performed to verify attainment of fatigue and damage tolerance design requirements for the composite wing box. A full-scale composite wing box semispan will be subjected to design service loads spectra equivalent to two aircraft service lifetimes to identify critical areas of the composite wing box not previously identified by analysis or component tests and to provide a basis for service inspection intervals and repair procedures. The test article will consist of a composite wing box semispan, an aluminum wing box semispan, and a center fuselage section. The composite wing box will be representative of the flight article. It can utilize the same test fixtures and equipment as the static test with modifications as required for any items that are utilized only for fatigue and damage tolerance tests. The test sequence for this test is shown in Figure 7-24.

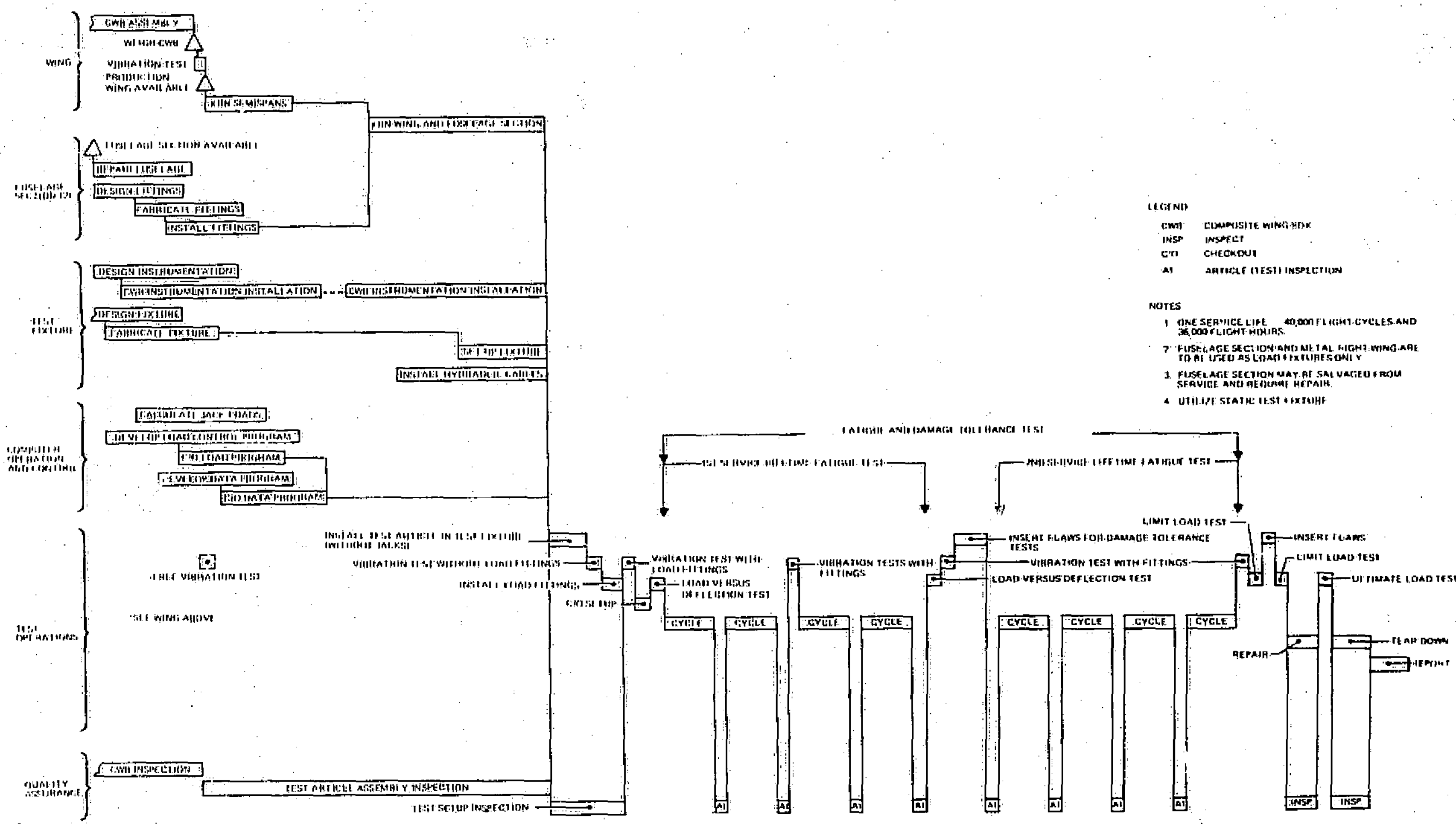


FIGURE 7-24. COMPOSITE WING FATIGUE AND TOLERANCE TEST PROGRAM

Detail test conditions and loads are to be provided by Design Engineering for the tests noted in Table 7-13. Loads will be defined based on DC-9 Series 30 specifications. Loads spectra will be flight-by-flight, with random odering of both flights and load peaks and valleys. Simplified profiles will be developed from typical service operations. Condensed spectra will be used whenever possible to reduce computer time and test costs. These spectra will be prepared and submitted for FAA approval.

TABLE 7-13
COMPOSITE WING FATIGUE AND DAMAGE TOLERANCE TEST REQUIREMENTS

ORIGINAL PAGE IS OF POOR QUALITY

TEST NO.	TYPE TEST	TEST LOCATION	PURPOSE	LOADING	LOAD LEVEL OR TYPE	ENVIRON	INSTRUMENTATION	REMARKS
1	FREE FREE VIBRATION (CWB SEMISPAN)	MANUFACTURING		SEE TEXT				
2	INSTALLED CWB VIBRATION TESTS A. PREFATIGUE TEST WITHOUT LOAD FITTINGS B. PREFATIGUE TEST WITH LOAD FITTINGS C. HALF LIFE CYCLE (20K LANDINGS) TEST D. LIFE CYCLE (40K LANDINGS) TEST E. TWO LIFE CYCLES (80K LANDINGS) TEST	STRUCTURAL TEST LAB		SEE TEXT				
3	LIMIT STATIC LOAD A. PREFATIGUE TEST		OBTAIN BASE-LINE DEFLECTION DATA	POSITIVE 100% DLL	100% DLL		DEFLECTOMETERS AND STRAIN GAGES	PRE- AND POST DATA TO BE COMPARED TO DETERMINE IF STIFFNESS IS ALTERED BY TWO SERVICE LIFETIMES OF LOAD APPLICATION.
	B. POST DAMAGE TOLERANCE TEST		OBTAIN POST CYCLIC TEST DEFLECTION DATA					
4	FATIGUE TEST		EVALUATE FATIGUE LIFE	ONE SERVICE LIFE TIMES OF LOAD APPLICATION	GROUND-AIR-GROUND LOAD SPECTRA		AXIAL (30), SHEAR (30), 2-GAGE ROSETTES (60), AND 3-GAGE ROSETTES (60); STRAIN GAGES; LOAD CELLS	
5	DAMAGE TOLERANCE TEST(S) A. FULL-SCALE CWB		EVALUATE DAMAGE TOLERANCE DURING FATIGUE CYCLING	AS FOR TEST NO. 4	AS FOR TEST NO. 4		AS FOR TEST NO. 4	MONITOR ANY CRACK GROWTH DURING TWO LIFETIMES. EVALUATE REPAIR DAMAGE TOLERANCE
	B. CWB COUPON TESTS	PHYSICAL TEST LABS	EVALUATE DAMAGE TOLERANCE OF COUPONS AND COMPONENT SEGMENTS FROM CWB FATIGUE TEST ARTICLE	100% DLL	SPECTRUM LOADING		TO BE DEFINED PER INDIVIDUAL TEST SPECIMEN	COUPONS/PARTS REMOVED DURING FATIGUE TEST ARTICLE TEARDOWN AND INSPECTION
6	FAILURE TEST	STRUCTURAL TEST LAB	DEMONSTRATE CRASHWORTHINESS	MAIN GEAR BREAKAWAY	FAILURE LOAD FOR MAIN GEAR BREAKAWAY		STRAIN GAGES (TBD), LOAD CELLS (TBD), MOVIE FILM	TEST REQ'D ONLY IF GEAR ATTACHMENT IS SIGNIFICANTLY REDESIGNED AFTER TEST OF MLG BREAKAWAY

Instrumentation will consist of deflectometers, strain gages, load cells, pressure transducers and associated signal conditioning, calibration equipment, power supplies, cabling, oscilloscopes, and other instruments to obtain structural response and for correlation to analysis. The fatigue test article will also be used to obtain vibration test data.

A stiffness test is to be conducted as the first installed structural test on the test article. Loads will be applied to the wing to 100 percent design limit load to measure wing bending and torsional deflection prior to fatigue testing. A second wing stiffness test will be conducted in a like manner after the first lifetime of cyclic loading is completed and a third test at the end of the second lifetime of cyclic loading. Data from these tests will be compared with the deflection data obtained from the first test to determine if the composite wing box deflection characteristics change as a function of loading and aging during the test.

Cyclic loads are to be applied as flight-by-flight spectra for fatigue and damage tolerance evaluation. Each service life will be divided into periods (cycle) with visual and nondestructive inspection of the complete test article at the end of each period. Jacks and whiffletrees will be disconnected at half of the first lifetime and a second vibration test conducted and at the end of the first lifetime. Significant damage which would result in premature test article failure will then be repaired. Flaws will be introduced in selected critical areas, if none exist, for the second service life to evaluate flaw growth characteristics and prove the structure is damage-tolerant. Load equipment will be reattached and cyclic tests continued for a second service lifetime in the same manner as for the first lifetime except that more local inspections are anticipated to monitor flaw growth in damaged area.

A fail-safe limit load condition will be applied after completion of the second service life of load cycling. Flaws will then be placed in selected undamaged areas and limit load applied. Damage will then be repaired, where practical, and tests conducted to ultimate design load to prove the capability to repair major damage to the composite wing box structure.

Full-Scale Wing Design Verification Modal Vibration Tests — Design Verification modal vibration tests are to be performed using the three full-scale semispan composite wing box test articles. Tests are to be conducted (1) on all semispan components immediately after assembly and before joining with the aluminum wing box, (2) on the two assembled ground test composite wing box test articles, and (3) on the composite wing box installed as the left wing of the flight test aircraft. These tests are to be performed to (1) determine normal composite wing box semispan basic vibration characteristics — mode shapes, frequencies, damping, and linearity, (2) obtain vibration data indicative of manufacturing reproducibility, (3) obtain vibration data for evaluation of possible structural degradation as a result of fatigue testing, (4) obtain data for correlation with modal vibration analyses, and (5) provide data necessary in demonstrating that the aircraft with a semispan composite wing box installed has flutter and vibration characteristics acceptable for aircraft flight. Table 7-14 summarizes all design verification vibration tests. Test No. 9 of the table is discussed separately under Structural and Aerodynamic Damping Tests.

A free/free pendulum type vibration test fixture will be provided, essentially as shown in Figure 7-25. This fixture will be used for Tests No. 1, 2, and 8 of Table 7-14. Data from the free/free tests performed on the three test articles will be correlated with dynamic analyses as an indicator of manufacturing reproducibility.

Vibration tests will be conducted on the assembled composite wing box as part of the complete test article, as shown in Figure 7-21. Five tests will be performed on the fatigue test article. Tests will be conducted to check for possible changes in frequency response of the composite wing box as a result of fatigue cyclic testing. Changes could be indicative of degradation of the test article as a result of load cycling or aging. All of these tests will be conducted in the same manner. Reference accelerometers will be installed at locations on the composite wing box upper surface. Roving accelerometers will be used to measure the modes of vibration. Shakers will be installed under the wing and attached with suction pads to excite the wing in bending and torsion. Mode shape, frequency, and damping data will then be obtained for three modes. Test results will be correlated with design analysis data after accounting for the jack pads attached to the composite wing box skin.

TABLE 7-14
COMPOSITE WING BOX DESIGN VERIFICATION VIBRATION TESTS

TEST ARTICLE	TEST NO.	TYPE TEST	PURPOSE	ARTICLE SUPPORT	TIME PHASING	TEST CONDITIONS	DATA REQUIRED	REMARKS
FULL SCALE STATIC TEST ARTICLE	1	FREE/FREE (CWB SEMISPAN)	DETERMINE NORMAL MODES OF VIBRATION, FREQUENCIES	BUNGEE	PRIOR TO WING JOINING	5 TO 50 Hz EXCITATION	VIBRATION MODES WITH FREQUENCY AND DAMPING	FREE, UNJOINED STRUCTURE IN CENTER TANK AREA MUST BE RESTRAINED (TYPICAL FOR SEMISPAN TESTS)
FULL SCALE FATIGUE AND DAMAGE TOLERANCE TEST	2	FREE/FREE (CWB SEMISPAN)	COMPARE RESULTS WITH TEST NO. 1	SAME AS FOR TEST NO. 1				CORRELATION OF TEST NO. 1 AND 2 RESULTS IS INDICATIVE OF PRODUCTION REPRODUCIBILITY
	3	ASSEMBLED TEST ARTICLE, CANTILEVERED	COMPARE RESULTS WITH TEST NO. 2	TEST FIXTURE	PRIOR TO LOAD PAD INSTALLATIONS	2 TO 50 Hz EXCITATION		REFERENCE LEVEL VIBRATION DATA FOR COMPARISON WITH DATA FROM TESTS 6, 7, AND 8
	4	SAME AS TEST NO. 3 WITH LOAD PADS	OBTAIN FATIGUE AND DAMAGE TOLERANCE TEST BASELINE VIBRATION DATA		AFTER LOAD PAD INSTALLATIONS, PRIOR TO LOAD VERSUS DEFLECTION TESTS	SAME AS FOR TEST NO. 3		EVALUATE DATA FOR VIBRATION CHARACTERISTIC CHANGES AFTER NOTED HOURS OF LOAD CYCLING. CHANGES IN CHARACTERISTICS MAY BE INDICATIVE OF LOSS OF RIGIDITY DUE TO AGING AND FATIGUE OF FIBERS
	5		OBTAIN VIBRATION DATA AFTER ONE-HALF SERVICE LIFE OF STRUCTURAL LOAD CYCLING		AFTER 20,000 HOURS OF LOAD CYCLING			
	6		OBTAIN VIBRATION DATA AFTER ONE SERVICE LIFE OF LOAD CYCLING		AFTER 40,000 HOURS OF LOAD CYCLING			
	7		OBTAIN VIBRATION DATA AFTER TWO SERVICE LIVES OF LOAD CYCLING		AFTER 80,000 HOURS OF LOAD CYCLING			
FLIGHT TEST AIRCRAFT	8	FREE/FREE (CWB SEMISPAN)	SAME AS FOR TEST NO. 1					CORRELATE RESULTS WITH TESTS NO. 1 AND 3
	9	AIRCRAFT GROUND VIBRATION TEST	OBTAIN SYMMETRIC AND ANTISYMMETRIC VIBRATION MODES, FREQUENCIES, AND DAMPING DATA FOR FINAL, COMPLETE AIRCRAFT	AIRCRAFT ON SOFT BUNGEE SUPPORTS	PRIOR TO FIRST FLIGHT	ZERO, HALF, AND FULL FUEL	MODES WITH FREQUENCIES AND DAMPING	DATA TO BE CORRELATED WITH WING FLUTTER AND GUST ANALYSES

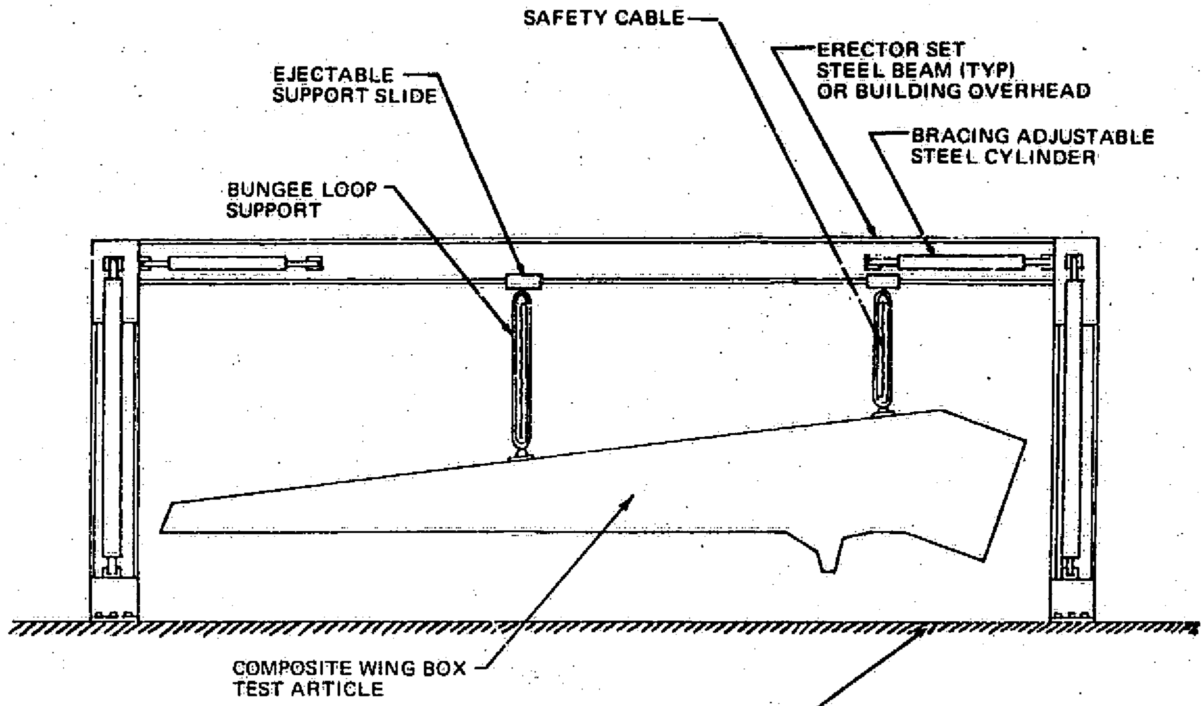


FIGURE 7-25. COMPOSITE WING BOX FREE/FREE VIBRATION TEST SETUP

Repair and Test of Major Damage – Repair of damage covers many areas of the composite wing box development program – repair of components damaged during manufacture, parts damaged during test, and parts damaged during in-service flight operation. Major test articles, including the crashworthiness box section and the two ground test composite wing box semispan test articles, will be repaired and tested where practical to develop approved techniques and procedures for in-service repairs. Through these repairs and tests, repair techniques will be developed for use on full-scale major test articles suitable for repairing parts during manufacture and aircraft in-service repairs and to demonstrate that the structural integrity of the repaired parts is equivalent to the original unrepaired structured. (See Reference 2, Para. 7.g)

Aircraft Installations Ground Tests

Certain ground tests are required on the flight test aircraft composite wing box and its subsystems during assembly or after installations are made on the flight test aircraft and before the first flight. These tests are performed in Phase V and are described in subsequent paragraphs.

Subsystem Functional Tests — System functional tests are performed during manufacture of an aircraft. These tests are to be accomplished with on-aircraft test procedures prepared by Engineering. Existing procedures will be used for system functional tests of the DC-9 produced with an advanced composite left wing box. These will be reviewed, modified as required, and utilized to demonstrate the satisfactory function of all systems in or interfacing with the left wing composite wing box. The following systems, in particular, will be examined closely to determine if changes are needed in procedures and during performance of the procedures for responses different than for a normal DC-9: (1) composite wing, fuel tank proof pressure and leakage test (2) electrical system (grounding and EMI), (3) flap system, (4) hydraulic system, (5) lateral control system, and (6) main landing gear.

No unusual conditions are anticipated for or in these system functions with the use of the composite wing box.

Fuel System Calibration and Gaging — A test will be performed to verify the physical and functional characteristics of the composite wing box portion of the aircraft fuel system before the first flight. It will be demonstrated that the composite wing box portion of the fuel system meets or exceeds the DC-9 fuel system containment and gaging specifications and FAR requirements. This test will be performed on the flight test aircraft.

Design analysis supplemented by computer programs is used to determine the fuel tank physical characteristics such as trapped fuel volume, tank expansion volume, high-level fill valve shutoff volume, and pump runout volumes. These fuel system characteristics as well as the accuracy of the fuel quantity gaging system will be confirmed by this test before the first flight of the test aircraft.

This test will be conducted after manufacturing acceptance and the dry functional testing of the fuel system and immediately before the ground vibration testing of the aircraft.

Lateral Control System Proof and Operation Test — A lateral control system proof and operation test will be performed to demonstrate the structural and functional integrity of the lateral control system installed in the composite wing before the first flight. A successful test will have been achieved when the lateral control system and associated support structure have sustained 100 percent design limit load with no permanent deformation and no slack cable fouling on adjacent structure. This test will be performed on the flight test aircraft assembled with a normal DC-9 right wing and an advanced composite left wing box. This test will simulate malfuctions and apply design limit loads to the cockpit controls. Data analysis and visual inspection of the lateral control system will substantiate the lateral control system structural and functional integrity as installed in the composite wing. These tests are to be conducted before the first flight.

Aircraft Ground Vibration Test — An aircraft ground vibration test will be performed to obtain structural normal modes of vibrations and corresponding frequencies and damping characteristics of the overall aircraft with a composite structure left wing and a normal DC-9 metal right wing. The test aircraft will be structurally complete with all major weight items included. Any major weight item missing is to be adequately simulated and installed in its proper position. Determination of important structural modes is required for evaluating flutter characteristics, gust analyses, and the airplane structural responses to ensure compliance with applicable FAA requirements. Detection and measurement of structural modes of vibration will be made up to 10 Hz with orthogonality within 10 percent.

The aircraft will be positioned for soft support with the landing gear suspended on bungees. It will be weighed and ballasted as required for each test configuration. Force shakers will be used to excite vibration, with the vibration sensed using accelerometers and the response data recorded.

Tests will be conducted with zero fuel, full fuel, and one intermediate fuel loading.

The aircraft ground vibration tests will be completed before the first flight.

Flight Test Program

The fourth test composite semispan wing box will be installed as the left-hand wing and will undergo flight testing after the ground vibration tests are completed and an experimental certificate has been received from the FAA. Details of the flight test program will be developed in accordance with the requirements discussed in the following text.

FAA Certification — The flight test demonstration of a semispan composite wing installed on a DC-9 aircraft will be limited to those items that could be affected by this change.

An FAA certification test requirement program will be prepared by the Douglas flight test engineers with coordination and agreement with FAA personnel to show compliance with pertinent FAA regulations.

Ground Test — A friction check of the lateral control system will be conducted to verify that any revised routing of lateral control system cables has not changed the established friction characteristics of the system.

Flying Qualities — Flying qualitites will be investigated to verify that the aeroelastic characteristics of the composite wing produce the same handling qualities as the conventional structure. The following tests will be qualitatively evaluated: (1) maneuvering stability — ± 0.5g increment, cruise configration; (2) roll rates — high speed; (3) static longitudinal stability — high speed, climb configuration, and (4) static lateral stability — cruise configuration.

Structural and Aerodynamic Damping Tests — Structural and aerodynamic damping tests will be conducted to ensure the composite wing is free from flutter and excessive vibration for all flight conditions to V_D/M_D. The aircraft will be tested at the most critical configurations as determined by analysis.

The aircraft will be tested at three discrete altitudes to V_D/M_D. The aileron damper on the composite wing will have rotational free play of ± 0.5 degree to simulate excessive wear cases. The aileron control surface on the composite wing will be mass-balanced to the critical end of limits established for flight test. (Flight test limits are more severe than in-service mass balance limits.) The wing structural modes will be excited by means of aileron, rudder, and elevator inputs consisting of two basic techniques, surface pulse and surface oscillation.

Instrumentation requirements for the structural and aerodynamic damping program are as follows:

- Left and right wing tip normal acceleration

- Left and right horizontal stabilizer tip normal acceleration
- Vertical stabilizer tip lateral acceleration
- Aircraft center-of-gravity normal acceleration
- Aircraft center-of-gravity lateral acceleration
- Cockpit normal acceleration
- Cockpit lateral acceleration
- Left-hand wing aileron position (assumes a composite wing)
- Left-hand elevator position
- Rudder position
- Captain's airspeed, altitude, and Mach number.

Flight Loads – A number of strain gages will be installed and monitored on various wing structural components throughout the flight test program. The data from these gages will be compared to analytical and static test data.

Airline Service – Following the flight test program and FAA certification, the flight test composite wing will be refurbished and the aircraft delivered to the user airline to begin in-service flight evaluation.

QUALITY ASSURANCE PLAN

Quality assurance activities will begin early in Phase I of the wing program to ensure that the design allows for access for inspection and to isolate areas of activity unique to this design. Quality control procedures will be written or revised to cover activities needed because of unique aspects of the wing. Inputs from the development programs will be incorporated at the earliest possible moment. Specifications will be revised as appropriate to cover material procurement and process control. These efforts will result in a comprehensive quality assurance plan tailored to the manufacture of the composite wing.

This Quality assurance Plan has been prepared as a guide to the quality assurance system to be used for the composite wing program.

The quality system is designed to conform to Part 21 of the Federal Aviation Regulations.

The Quality Assurance Plan provides the controls that will ensure quality and conformity of the composite wing components. Specialized controls are described for the production of the graphite/epoxy components.

The quality system incorporates all necessary controls for the effective assurance of product quality. The quality assurance management system provides senior management with visibility of the program. General quality information and records will be available to NASA representatives.

Specification Review, Inspection Review, Drawing and Change Control

The specification review is conducted by Design Engineering and the Materials and Producibility Engineering groups. Specifications are reviewed and, where applicable, are incorporated into engineering drawings and process standards.

Quality Assurance reviews preliminary design layouts and final design drawings to verify that all composite components are readily accessible for normal and nondestructive inspection.

Change control is maintained for produced hardware, including status of parts and assemblies adapted from other programs. Change control is effected by cognizant engineering and planning personnel. Engineering assigns a change letter to a drawing to identify the change, and this identity is included on the planning paper and the product. Quality Assurance verifies the change conformity as part of the hardware acceptance.

Processing Instructions, Material Specifications, and Quality Specifications

Douglas process standards convey detailed processing instructions, material usage, quality control procedures, and quality requirements used to manufacture and evaluate the quality of parts.

Material specifications document material handling, processing, and mechanical and physical properties of materials to ensure that materials used will produce hardware that meets design requirements. These specifications detail material properties as well as testing requirements which are imposed upon material suppliers.

Quality specifications are written to convey to both suppliers and the manufacturer the quality control information necessary for ensuring compliance with engineering, quality assurance, contractual, and regulatory requirements.

Process standards and material specifications are indicated on the engineering drawings as applicable. Process standards, material specifications, and quality specifcations are indicated on material purchase orders. All these items are specified on the manufacturing paper, as required.

Personnel Training and Qualification and Assignment of Quality and Inspection Stamps

Specially trained and qualified employes are required to perform highly technical work operations. Training courses are provided for employes involved in such work. Upon completion of training, the employes are given tests to demonstrate their proficiency in performing the task. Only employes who show proficiency and pass the qualification tests are used to manufacture and lay up composite parts.

Training is provided to Quality Assurance personnel for both technical and procedural subjects. Both classroom and on-the-job training are utilized. Courses are designed to ensure the technical competency of personnel and effective implementation of quality assurance system requirements. Where applicable, certifications to perform specific inspection operations are issued. Quality Assurance personnel directly assigned to product verification activities are issued inspection stamps.

Both the quality stamp and inspection stamp are uniquely designed to give traceability to the individual employe. A computerized system is used to provide traceability to the employe as well as certification status.

Supplier Evaluation

Prospective suppliers are evaluated to determine if they have acceptable quality assurance systems and capabilities.

Follow-up surveys are conducted as necessary to verify correction of deficiencies disclosed by initial surveys.

Postaward supplier surveys are conducted for corrective action purposes, as required.

Quality Planning

Quality planning ensures effective control of quality throughout procurement, fabrication, and assembly.

Purchase requisition analysis is performed to verify that adequate quality requirements are incorporated. This analysis includes a review of the applicable engineering requirements and quality specifications.

Inspection requirements are established for fabrication operations and the fabrication outline is checked to ensure adequate inspection operations are specified on the fabrication outline.

Assembly operations and associated inspection functions are planned and documented on assembly outline – ships record forms. Quality Assurance personnel coordinate with Manufacturing Planning to ensure that appropriate inspection operations are indicated on the assembly outline – ships record.

Process Control

Process surveillance is conducted by Quality Assurance – Process Control to ensure that product-related technical processes utilized at the manufacturing facilities comply with specifications. Corrective action is required for reported deficiencies.

Processing suppliers are qualified in accordance with quality specifications. Resurveys are made periodically of qualified sources.

Equipment Certification

Measuring and testing equipment used to ensure or verify product conformity is calibrated and ceritified for accuracy prior to its initial use and at prescribed intervals thereafter. Calibration of primary and secondary measurement standards used for equipment certification is traceable to the National Bureau of Standards.

Inspection and Tests

Inspections and tests are conducted to verify compliance of the product with specifications and procedures. Quality Assurance personnel witness all tests to ensure this compliance. Documented inspection and test results are traceable to personnel performing the function. Equipment utilized for testing is monitored to ensure that its control, maintenance, and calibration are in compliance with procedures and specifications.

Tooling Inspection

Tooling inspection ensures design conformity of product features controlled by production tooling. Tooling is visually inspected before and during use to detect any condition that may affect product conformity or acceptance. When required by Tool Design, periodic dimensional checks are made. The quality assurance record – tooling and the tooling order are utilized to specify and record tooling inspections.

Receiving Inspection

Receiving inspection of product materials is conducted in accordance with applicable material specifications and purchase orders. Visual inspections are conducted in the receiving area. Material requiring physical or chemical testing is routed to the appropriate laboratory for analysis.

Graphite/epoxy material procured for the program undergoes receiving inspection and acceptance testing by Quality Assurance to requirements set forth by Materials and Producibility Engineering.

Nonconforming materials are segregated and processed for disposition and corrective action in accordance with applicable control procedures. Accepted materials are routed to the appropriate cold storage, stockroom, or use area. Acceptance or rejection by Quality Assurance is documented on receiving documents.

Raw Material Control

Raw materials are purchased under applicable specifications and accompanied by documented certification when required by procurement documents. The results of tests (e.g., chemical or physical) conducted on raw material specimens are documented. Traceability of graphite/epoxy raw material is verified by Quality Assurance from material procurement through assembly as recorded on production work orders.

Raw materials that require cold storage are certified for a specific period of time. Materials that exceed this storage age are tagged and held for retest by the appropriate laboratory for recertification. Accepted materials are marked for a new specified time period.

Fabrication Inspection

Parts and assemblies manufactured in the fabrication areas are inspected in accordance with quality instructions as provided on the fabrication outlines. Quality and completeness of fabrication items are signified by the application of quality stamps by manufacturing personnel. Quality Assurance verifies that batch or lot number of graphite/epoxy material and serialization of components (as applicable) are recorded on the fabrication outline for traceability.

When specified on the fabrication outline tag end specimens are prepared and tested in accordance with engineering instructions and recorded. Quality Assurance visually inspects each layer of material prior to layup to verify proper orientation. Fabrication outlines provide buyoff for each layer.

The completed components are visually inspected for cracks, delaminations, and other flaws which are documented and dispositioned pursuant to directions from the Material Review Board. Ultrasonic and/or Fokker bondtest and radiographic inspection are performed by Quality Assurance per a detailed written procedure to detect delaminations. Unacceptable conditions are documented and dispositioned per the Material Review Board.

Assembly Inspection

Quality and completeness of assembly operations are signified by the application of manufacturing quality stamps to the assembly outline — ships records.

Quality Assurance personnel provide acceptance of subassemblies and final assemblies by applying a quality assurance inspection stamp on each item inspected. Quality Assurance verifies that serial numbers of the components are recorded on the assembly outline — ships records to ensure traceability, as required. Completed assembly outline — ships records are retained by Quality Assurance in data records.

Flight Ramp Inspection

Flight ramp inspection operations are performed by manufacturing personnel in parallel with preflight functions and production flight test support activities. Inspections are planned, conducted, and recorded as an extension of assembly inspection activities. Production flight testing is accomplished by the Flight and Laboratory Development organization.

Material Review

Product nonconformances are controlled to ensure their correction or disposition per the Material Review Board. Nonconforming manufactured items that can be corrected to comply with specifications are returned to Manufacturing for correction. Other nonconforming items are rejected and withheld for material review processing. The Material Review Board consists of a cognizant engineer, a cognizant Quality Assurance representative, and a cognizant Government representative (when required).

Corrective Action

Material Review Board actions and discrepancies disclosed by inspections and surveillance are analyzed. Corrective action is obtained as necessary. Follow-up measures provide for verification of effectiveness of reported corrections. In-service problems are reported to Quality Assurance by Product Support for in-house corrective action as required.

Quality Audits

Quality audits are conducted to verify compliance with established procedures and specifications and to identify any apparent deficiencies in the quality system. Detailed audit reports are furnished to affected subdivisions listing audit findings and any recommended actions to be taken. Responses are reviewed and follow-up audits conducted to verify the satisfactory resolution of reported deficiencies.

Inspection and Test Records and Data

Those records that provide objective evidence of assembly acceptance are retained by Quality Assurance. These records and supporting data provide documentation of inspection and test results.

PROGRAM COSTS

The determination of the cost of a composite wing technology program was not included in the study task. However, rough-order-of-magnitude costs were estimated in order to compare program options and to define a minimum-cost development plan without compromising the program objectives.

The development plan cost breakdown is presented in Figure 7-26 and Table 7-15 to provide insight into the scope of the various program tasks.

These cost data were not developed through the rigorous and lengthy bid-work sheet and costing department procedures, and therefore should not be construed as suitable for any other purpose.

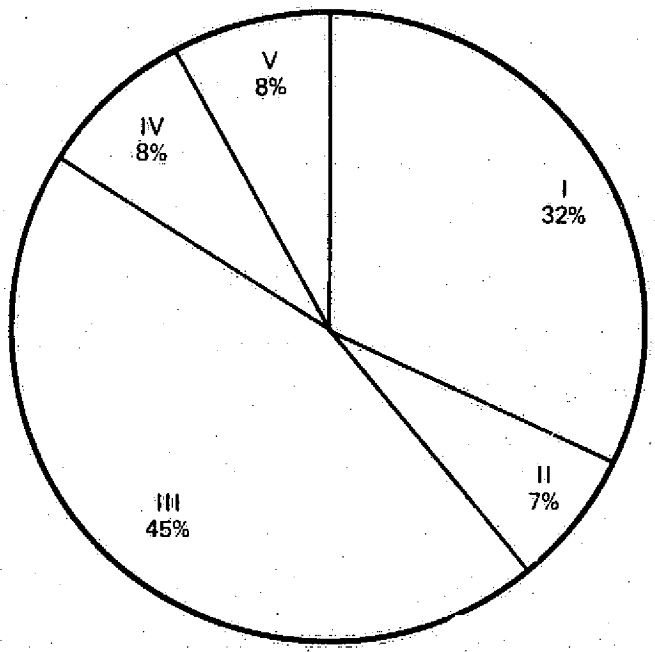


FIGURE 7-26. DISTRIBUTION OF FUNDS AMONG PHASES

TABLE 7-15
DEVELOPMENT PLAN COST SUMMARY

	APPROXIMATE 1978 DOLLARS (MILLIONS)
PHASE I	
ENGINEERING PRELIMINARY DESIGN	4.0
MANUFACTURING TECHNOLOGY DEVELOPMENT	11.5
DEVELOPMENT TESTING	10.1
PHASE II	
ENGINEERING DETAIL DESIGN	2.5
VERIFICATION TESTS	0.5
MANUFACTURING VERIFICATION	0.5
PHASE III	
TOOLING	29.4
MANUFACTURING (3CWB)	6.3
PHASE IV	
FULL-SCALE VERIFICATION TESTS	4.8
PHASE V	
MANUFACTURING (1 CWB)	2.1
GROUND TESTS	0.4
FLIGHT TESTING	1.0
AIRCRAFT MODIFICATIONS	1.3
ENGINEERING MODIFICATIONS	0.4
	74.8
TRAVEL, COMPUTER, MISCELLANEOUS	0.8
	75.6

SECTION 8
FACILITIES AND EQUIPMENT

The facilities section of this study is divided between the development and production programs for the purpose of furnishing an overview of what is required during each phase. Basically, the facilities requirement for the development phase of composite wing manufacture will not differ greatly from the production phase except for the additional space and equipment necessary to achieve the production rate.

A plant design and work flow for structures laminated from composite materials is substantially different from those used in the metallic producing facilities of today's airframe manufacturers. This report is to define the facilities requirement for each phase of the composite wing manufacturing program.

A one-of-a-kind approach is used in development for manufacture of the four full-size composite wing box structures. The primary purposes are to develop technology, train personnel, and acquire a manufacturing capability for large primary composite structures for a reasonable capital outlay. In the real-world situation, the development program would be conducted in R&D facilities at Douglas which would be available in that time period. The only new capital expenditures would be for equipment which was large enough to manufacture the wing panels and spars. Hand operations would be utilized except where technical innovation required study for development in these areas. In addition, the basic R&D effort would require a study of handling and processing techniques necessary to manufacture composite wings on a production basis.

The facilities for the production program must include the latest techniques available to make composite wing production cost-competitive with metallic components. Hand layup will give way to automated broad goods dispensing, as well as numerically controlled trim equipment.

The key to composite wing production will be not only the producibility of the design, but also the methods and equipment used to manufacture the wings at a sustained rate of production.

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The production study will examine the manufacture of an aircraft wing in the 1985-1990 time frame, in which much of the equipment necessary to produce other primary and secondary structures would be available.

Facility and equipment requirements for the composite wing development tests and full-scale verification tests are essentially the same as those required for development of a metal wing of similar size. No unique requirements are foreseen at this time.

FACILITY REQUIREMENTS FOR DEVELOPMENT

This section defines the facilities necessary to produce one major subcomponent and four full-sized composite wing structures.

The facility will be based on a one-of-a-kind approach with the wing-half space envelope utilizing a DC-9 size aircraft. The objective is to develop methods applicable to production.

The equipment required during the development program will be of the type and size needed during production and would later be used in a production facility. Hand layup and trim would be utilized, where possible, to minimize expenditures for capital equipment. However, the composite wing development program will require that the development facility have sufficient capacity in the following areas: storage, layup, cure, nondestructive testing, trim, final assembly, portable hand tools and tooling support, surface treatment for metallic parts, and refrigerated storage area for work-in-process storage. Sufficient freezer capacity is necessary to accommodate the quantity of prepreg material to be used during the development phase with two wing-halves in process at any one time.

Material to be used in the development program will be kept separate from materials used in any production application.

Movement of completed major subassemblies will be via handling dollies until the postcure state where the overhead crane is available for transfer of large parts.

Initial fabrication of composite parts occurs in the layup room with the dispensing of tape and broad goods onto the skin molds, blade stringer mandrels, intercostal mandrels, and rib molds. After metallic inserts have been made, parts are densified and returned to storage or final assembly from the -10°C (0°F) freezer or 4°C (40°F) holding room storage, depending on length of time to be stored.

Final assembly is performed before vacuum bagging and final cure in an oven or autoclave. Postcure operations include removal from the tool, nondestructive testing, and trim using diamond saws and high-speed routers. The wing box will be assembled in a similar manner to its metallic counterpart, utilizing a vertical assembly jig and typical plant air and electricity for handheld tools.

The approximate overall size of a development wing is estimated to be 10,400 square meters (112,000 square feet) including area allocated to metallic part surface preparation, tool fabrication, and staging and receiving.

The nondestructive testing of the full-size development wing will require inspection capability to handle at least a 3.5- by 18-meter (12- by 60-foot) wing panel. Automated head control will be used for inspection of cured panels using the C-scan technique. The layup area for graphite/epoxy laminates will require temperature and humidity controls to maintain 10° to 24°C (65° to 75°F) at 50 to 70 percent relative humidity. Dust will be controlled by layup room positive air pressure. Sealed floors which are waxed frequently will reduce the accumulation of dirt and dust. Shop areas used for trimming cured composite parts will be lower than atmospheric pressure to prevent dust from infiltrating into the layup area. Both vacuum pickups and electrostatic collection systems can be used.

The plant layout design, shown in Figure 8-1, indicates the facilities required for production of the full-sized development composite wing. The basic philosophy is to provide an efficient work flow incorporating process flow techniques applied to production, but minimizing expenditures for capital equipment.

The basic elements of composite manufacture, layup, cure, and trim will be controlled in accordance with production specifications. The shop layout allows for prepreg material storage in freezers kept at -18°C (0°F) and metallic parts storage in a stockroom adjacent to the incoming receiving

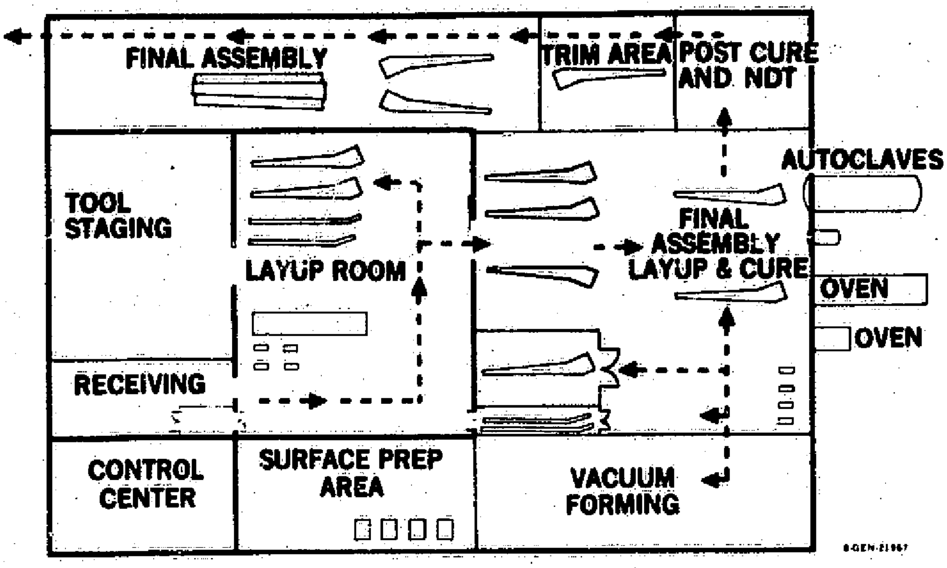

FIGURE 8-1. COMPOSITE WING DEVELOPMENT FACILITY

area. From the raw material storage area, the broad goods and tape are diverted to the layup area in quantities to be used in one application. Automatic dispensing and trim equipment will need to be used to dispense broad goods in quantity, not only for reasons of physical size but also because of the repetition required to handle the number of plies in each wing panel. A programmable waterjet could be used on a Gerber table for this task. The smaller subassemblies could be handled in a similar manner with shipsets of ribs and spars layed up in advance of wing panels and stored until needed. Each subassembly will be bagged in a silicone rubber blanket and tested for leaks before it is cured.

The wing panels will then be moved into the oven for densification as are the spars, ribs, and mandrels utilized for the intercostal buildup. Upon removal, the wing panels will either be returned to the layup room for final layup prior to final cure or stored in the holding room until needed. The holding room will be chilled to 4°C (40°F) to maintain or prevent premature curing. Freezer storage will be required for subassemblies to be stored for longer periods of time, including intercostal block mandrels and blade stringers.

Metallic parts to be bonded and cocured at the time of final assembly in the autoclave will undergo cleaning in the surface preparation area. Processing requirements will include Pasa-Jell, vapor honing, and vapor degreasing for parts up to 1 by 2 meters (4 by 7 feet) in size.

Nondestructive testing of all subassemblies will occur immediately after final cure and removal of the part from the tool. Ultrasonic testing C-scan will be done using squirters and catch-pans. In-motion x-rays will also be necessary.

Final trim of all cured composites after nondestructive testing will be done in a controlled area with a ventilation system utilizing lower than atmospheric pressure and electrostatic filtration to collect dust particles. Typical trimming equipment in this area will include a diamond saw, track-mounted router, or tracer router tooling.

The final assembly area for the development program will allow for a vertical assembly jig and sufficient area for laydown of panels and spars before and during assembly. Storage space for subassemblies and personnel equipment will be allocated. High-speed drills to be used in fastening all subassemblies will require a vacuum collection system for dust particles.

The building design will also provide the full crane coverage to handle movement of completed wing halves into shipping bucks and panels and spars from work station to work station.

The basic philosophy of the development program would require a minimum-risk approach to setting up a Wing Development Composite Facility. The probable real-world situation in the 1984-1990 time period would warrant the inclusion of the development wing program into a composite manufacturing facility which would be in operation at that time rather than the establishment of such a facility with redundant expenditure and duplication of the same equipment by Douglas.

The development plan facility requirements and an approximation of their cost are summarized in Table 8-1.

FACILITY REQUIREMENTS FOR A PRODUCTION WING

A facilities forecast for a production wing program should start with the premise that the utilization of composite primary wing structure will be preceded by the utilization of secondary and medium primary structure throughout the airframe.

Table 8-2 shows a road map for McDonnell Douglas future plans for the utilization of composite structures through 1990. This road map is the basis for the facilities forecast shown in Figure 8-2. A normal production rate of one aircraft per week was assumed.

TABLE 8-1
DEVELOPMENT FACILITIES COMPOSITE WING

	APPROXIMATE COST 1978 DOLLARS (1000)
BUILDING STRUCTURES 9,290 m^2 (100,000 FT^2)	6,000
AUTOCLAVES	3,220
OVENS	195
FREEZERS	195
NONDESTRUCTIVE TESTING EQUIPMENT	620
IN-MOTION X-RAY	
ULTRASONIC C-SCAN	
WATERJET CUTTER ON GERBER X-Y TABLE	789
METAL PREP EQUIPMENT	175
VAPOR DEGREASER	
HONING EQUIPMENT	
METAL CLEANING LINE	
PASA-JELL	
TRIM EQUIPMENT	129
MACHINE SHOP	60
TOTAL	11,383

TABLE 8-2
COMPOSITE APPLICATIONS ROAD MAP

<table>
<tr><th></th><th>DC-10 STRETCH</th><th>ADVANCED TECHNOLOGY AIRCRAFT</th><th>SHORT HAUL</th></tr>
<tr><td>PROGRAM ATP</td><td>1979</td><td>1984</td><td>1988</td></tr>
<tr><td>1ST DELIVERY</td><td>1982</td><td>1987</td><td>1991</td></tr>
<tr><td rowspan="3">COMPOSITE SECONDARY STRUCTURE</td><td>AFT RUDDERS</td><td colspan="2">CONTROL SURFACES NLG DOOR</td></tr>
<tr><td colspan="3">TRAILING EDGE PANELS
FAIRINGS
LONG DUCT NACELLE</td></tr>
<tr><td></td><td colspan="2">FLAPS GEAR DOORS
AFT FUSELAGE SECTION</td></tr>
<tr><td rowspan="3">COMPOSITE PRIMARY STRUCTURE</td><td colspan="3">FLOOR BEAMS AND STRUTS</td></tr>
<tr><td rowspan="2"></td><td colspan="2">VERTICAL STABILIZER</td></tr>
<tr><td></td><td>HORIZONTAL STABILIZER
WING</td></tr>
</table>

In addition to the overall forecast, an analysis was made which considers only the production of the primary wing box. Basic requirements for this facility are shown in Table 8-3.

Duplicate machines and equipment would be required where the capacity of equipment used for development would not permit the attainment of the assumed 1.0 per week rate and there automation can be used in place of hand layup. Also, peculiar processing equipment of the type and size necessary to cure a full-size DC-9 wing would be procured to accommodate the larger production wings; e.g., short-haul wings, 4.5 by 21 meters (15 by 70 feet), would require at least a 6- by 23-meter (20- by 75-foot) clear working area in an autoclave and oven.

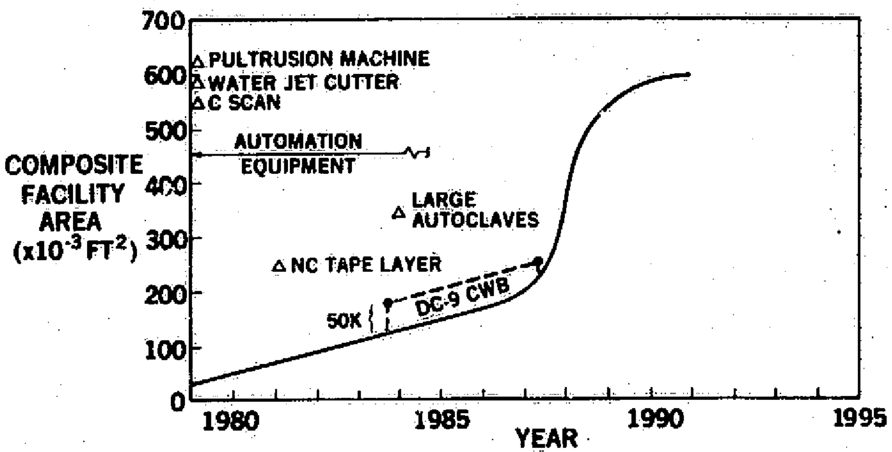

FIGURE 8-2. FACILITY FORECAST

TABLE 8-3

FACILITIES/EQUIPMENT REQUIREMENTS

	DEVELOPMENT FACILITY	PRODUCTION FACILITY
TOOLS	1 SET	4 SETS
CUTTING EQUIPMENT	WATERJET SYSTEM	2 WATERJET SYSTEMS
NDI	C-SCAN X-RAY	ADDITIONAL C-SCAN X-RAY
AREA	9,290 m^2 (100,000 FT^2)	13,935m^2 (150,000 FT^2)
MATERIAL STORAGE	2 FREEZERS	4 FREEZERS
CURING EQUIPMENT	2 AUTOCLAVES 2 OVENS	MULTI-SHIFT WORK CYCLE

8-GEN-21965
8-GEN-21966

Receiving and shipping of all composite materials will be handled and stored in accordance with applicable Douglas process specifications.

Metallic parts will be stored in a stockroom adjacent to the receiving area.

Two $-18^{\circ}C$ ($0^{\circ}F$) freezers, 3.5 by 9 by 3 meters (12 by 30 by 10 feet) high will be utilized for storing incoming prepreg material. One 6- by 21- by 3-meter (20- by 70- by 10-foot) high $-18^{\circ}C$ ($0^{\circ}F$) freezer will be located between the layup and final assembly/cure areas for storage of in-process subassemblies.

Overhead crane coverage will be provided throughout the assembly areas to handle movement of wing panels and completed wing box assemblies.

Fabrication of wing box subskins, stringers intercostals, and ribs will be accomplished in the production facility similar to the development facility shown in Figure 8-1.

The same nondestructive testing equipment will be used for quality assurance as used in the development phase.

Among considerations for the production facility should be the installation of auxiliary power systems to minimize the potential loss of power to all processing, storage, and curing equipment, resulting in loss of all prepreg material.

All layup and trim areas will have environmentally controlled atmospheres as required by Douglas process standards and industrial standards for the control of toxic particles. Specifically, all layup areas will have dust-free work areas with positive pressure ventilation systems. Humidity and temperature controls are necessary in all layup areas with specifications of 18° to $27^{\circ}C \pm 3^{\circ}C$ (65° to $80^{\circ}F \pm 5^{\circ}F$) temperature and relative humidity controls between 50 and 70 percent.

Trim areas require a slight negative pressure and electrostatic dust-collection system.

Finally, housekeeping requirements should be of such a nature as to provide a clean room environment conducive to a good working layup and cure area.

SECTION 9
APPLICATION AND BENEFITS

The road map presented in Table 8-2 reveals McDonnell Douglas Corporation plans for extensive applications of advanced composite materials in future commercial transport aircraft.

The short-haul transport scheduled for introduction in the late 1980s has been selected as the most timely vehicle for composite primary wing structure. As indicated by the road map, extensive applications of secondary and medium primary composite structures will precede the introduction of primary wing structure.

The short-haul transport configuration data include two or four wing-mounted engines depending on the type of engine, an operator's empty weight of 58,060 kg (128,000 pounds), a payload of 16,459 kg (36,285 pounds), 177 single-class passengers, and a range of 6241 kilometers (3370 nautical miles).

Figure 9-1 depicts the best-estimate schedule relationship between the composite wing technology program and the introduction of the short-haul transport aircraft. In early 1980 to 1984, a management decision must be made in order to market the short-haul transport with a composite wing structure and to develop the advanced design to the level necessary to complete the detail design fabrication and assembly within approximately 19 months following a production go-ahead decision. This decision will have to be made on the basis of Phase I technology and data acquisitions, supported by a firm commitment that the other five phases will be carried out.

An analysis was made to determine fuel savings of the short-haul transport with advanced composite structure over conventional aluminum structure in accordance with the road map. The analysis does not include any resizing of the aircraft or engine changes to account for the reduced structural weight. The results of the analysis are shown in Table 9-1. The total weight saving estimate of 4445 kg (9800 pounds) was derived from in-house experience with secondary structure, proven results from the NASA ACEE DC-10 composite rudder program, preliminary findings from the NASA ACEE DC-10 composite vertical stabilizer program, and the 28-percent weight saving reported herein for composite wing structure.

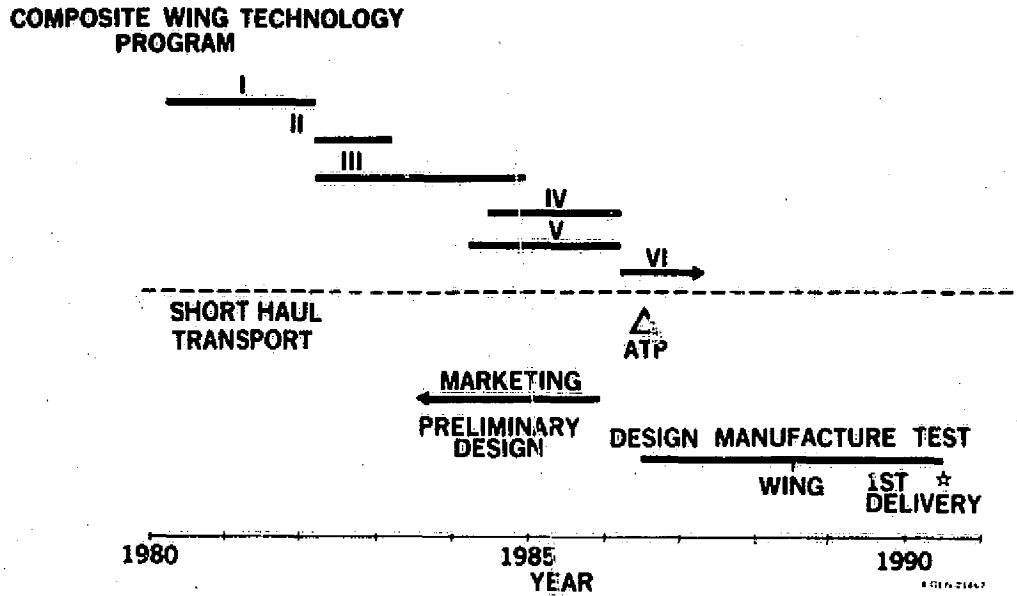

FIGURE 9-1. TIMELINES OF COMPOSITE APPLICATIONS TO SHORT-HAUL TRANSPORT

TABLE 9-1

COMPOSITE BENEFITS TO SHORT-HAUL TRANSPORT

WEIGHT SAVINGS – 4,445 kg (9,800 LB)

FUEL SAVINGS – 7,690,000 LITERS (2,000,000 GALLONS)

COST SAVINGS (BASED ON 20-YEAR LIFE)

- 1977 FUEL COST – $769,000
- PROJECTED AVERAGE COST – $2,600,000

OR

PAYLOAD INCREASE OF 4,445 kg (9,800 LB)

SECTION 10
STUDY CONCLUSIONS

The study supports the conclusion that a composite wing technology program must be undertaken by the commercial transport manufacturer to accomplish the transition from materials and practices utilized in current construction to extensive use of composites in wings of aircraft that will enter service around 1990. Data have been developed to define such a program.

The list of acceptance factors compiled for the manufacturer, FAA, and airlines provides a rational basis for an assessment of composite wing technology.

The assessment indicates the need for a composite wing technology program which contains the following provisions:

1. Development of technology and data to resolve the eight key issues defined herein.

2. Design, manufacture, and test of flightworthy, certifiable, full-scale hardware encompassing a range of wing design features representative of commercial transport aircraft.

3. Demonstration of composite wing technology to the extent that technical, economic, operational, and programmatic risks are reduced to an acceptable level.

4. In-service flight evaluation to provide realism to other phases of the program, and to demonstrate the operational performance of primary composite wing structure.

The conceptual design indicates that the goal of a 25- to 30-percent weight saving is attainable for primary composite wing structure compared to conventional aluminum structure, subject to further limitations which may be imposed as the eight key issues described herein are resolved.

A facilities and equipment plan should be prepared with the realization that the production utilization of composite primary wing structure will be preceded by extensive utilization of composite secondary and medium primary structure.

The utilization of composite primary wing structure in the 20th century on commercial transport aircraft is dependent on the continued NASA sponsorship of a composite wing technology program.

SECTION 11
STUDY RECOMMENDATIONS

1. A NASA-funded composite wing technology program is recommended to exploit the potential of using advanced composite materials for aircraft wings to provide a 25-percent weight saving with a promise of reduced costs throughout the life of the aircraft. These advantages can be realized as experience and technology accrue and mass production reduces material and manufacturing costs.

2. Critical path technology programs should be funded as soon as possible if the 1985-1990 goal for the introduction of primary composite wing structure on new aircraft is to be realized. The key issues which should be addressed first to supply data and technology in a timely fashion are:

   A. Repair of major damage

   B. Impact damage (included in durability issue)

   C. Damage tolerance design studies and tests

   D. Innovative molding methods

   E. Tooling methods for large composite structures

   F. Lightning protection.

3. The remainder of the durability key issues and the two other key issues of crashworthiness and NDI method can be started later in Phase I since sufficient basic data for these technologies are available to support early preliminary design tasks.

4. The NASA Fiscal 1979 budget should include funding to initiate contracted technology development programs with more than one airframe manufacturer for application to composite wing structure.

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