OPPORTUNITIES FOR COMPOSITES IN COMMERCIAL TRANSPORT STRUCTURES

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ACEE COMPOSITES PROGRAM-

INTRODUCTION

In recent years graphite/epoxy material has found widespread application in military aircraft and is now finding application in commercial transports. Because of special features of this material, such as high strength-to-density ratio, good formability and laminate tailoring, the next generation of military and commercial aircraft manufactured with composites could be significantly more efficient than current aircraft. Studies have shown that composites can reduce the structural weight of transport aircraft by as much as 25 percent over current aluminium structures with a corresponding reduction in fuel consumption of 12 to 15 percent. The NASA Aircraft Energy Efficiency (ACEE) composites program was established to foster the application of composite material in the next generation of aircraft. The primary objective of ACEE is to develop the essential technologies in cooperation with the commercial transport manufacturers to permit the efficient utilization of composites in airframe structure of future transport aircraft.

Specifically the ACEE composites program is to provide each of the commercial transport manufacturers both the technology and confidence required for a commitment to composite structures production. This means not only know-how for predictable designs and low-cost fabrication, but enough test and actual manufacturing experience to accurately predict durability for product warranty purposes and costs for product pricing, and to assure safety for certification by the FAA and maintainability for acceptance by the airlines.

OBJECTIVE

PROVIDE THE **TECHNOLOGY** AND **CONFIDENCE** SO THAT COMMERCIAL TRANSPORT MANUFACTURERS CAN COMMIT TO PRODUCTION OF COMPOSITES IN THEIR FUTURE AIRCRAFT:

SECONDARY STRUCTURE - 1980 TO 1985

PRIMARY STRUCTURE - 1985 - 1990

TECHNOLOGY

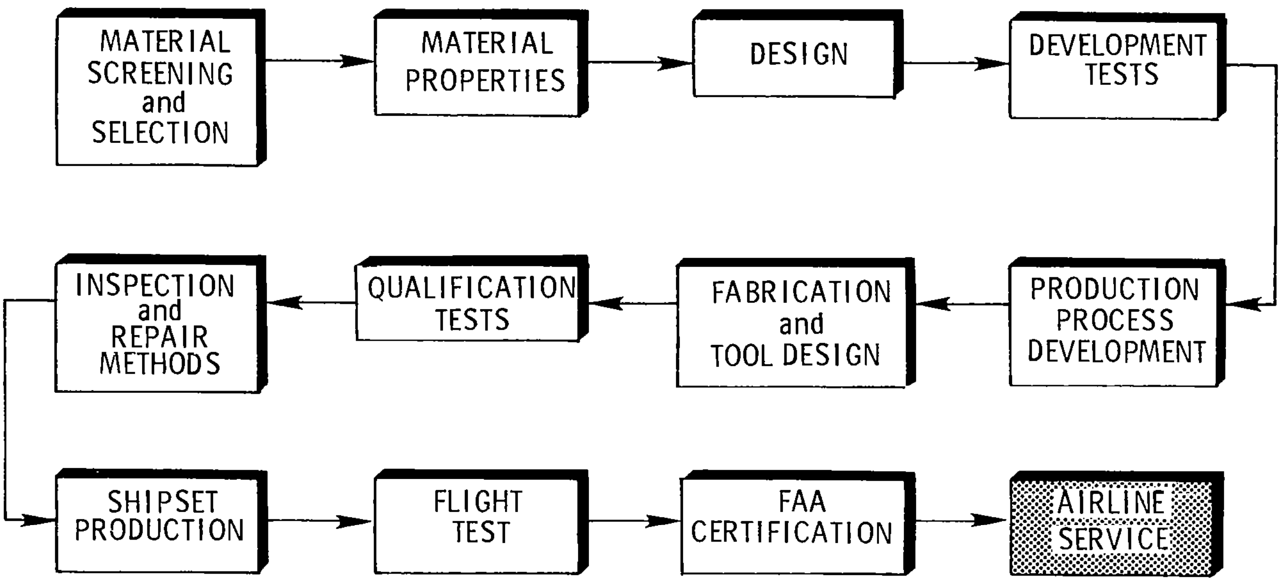
- DESIGN CRITERIA, METHODS AND DATA
- QUALIFIED DESIGN CONCEPTS
- COST COMPETITIVE MANUFACTURING PROCESSES

CONFIDENCE

- DURABILITY/WARRANTY
- *QUANTITY COST VERIFICATION*
- FAA CERTIFICATION
- AIRLINE ACCEPTANCE

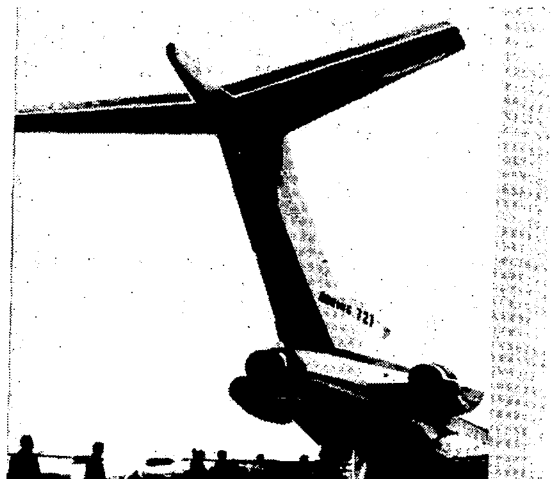
COMPONENT DEVELOPMENT TOWARD A PRODUCTION COMMITMENT

To support these objectives the manufacturers are developing composite versions of structural components on existing aircraft with NASA paying 90 percent of the cost. Development involves testing of various material options before selecting one and then extensive testing to develop an adequate data base of material strength and stiffness properties. Design options are narrowed through analysis and a varied spectrum of development tests on small and large subcomponents. In parallel with this, a suitable production process including economical ply preparation and cure at high temperature and pressure is evolved, tools are designed and fabricated, and full-scale components are then manufactured for ground qualification tests, flight tests, and airline service. The various tests include many that are required by the FAA for flight certification, which must precede airline service. Inspection and repair methods to insure adequate maintenance in service are also developed.

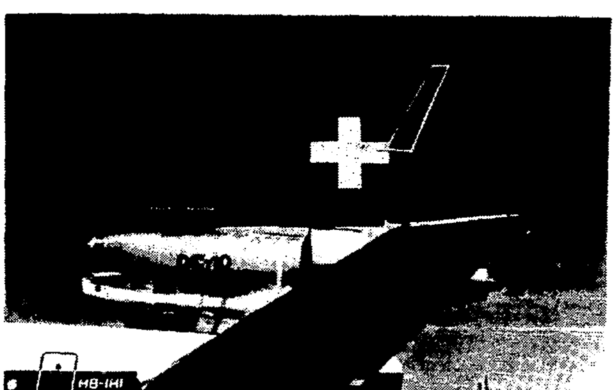


ACEE COMPOSITE SECONDARY STRUCTURES

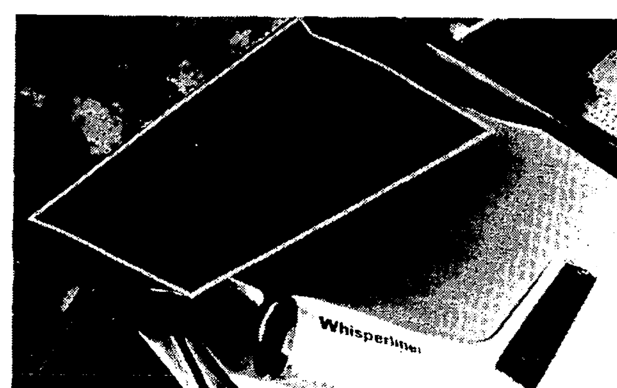
The commercial transport manufacturers were challenged to redesign selected secondary and medium primary components on existing aircraft with composite material. Secondary components include the Boeing 727 elevators, the Lockheed L-1011 ailerons and the Douglas DC-10 upper aft rudder. Such components are called "secondary" structures because they do not carry primary flight loads and are not critical to flight safety.



BOEING 727 COMPOSITE ELEVATOR



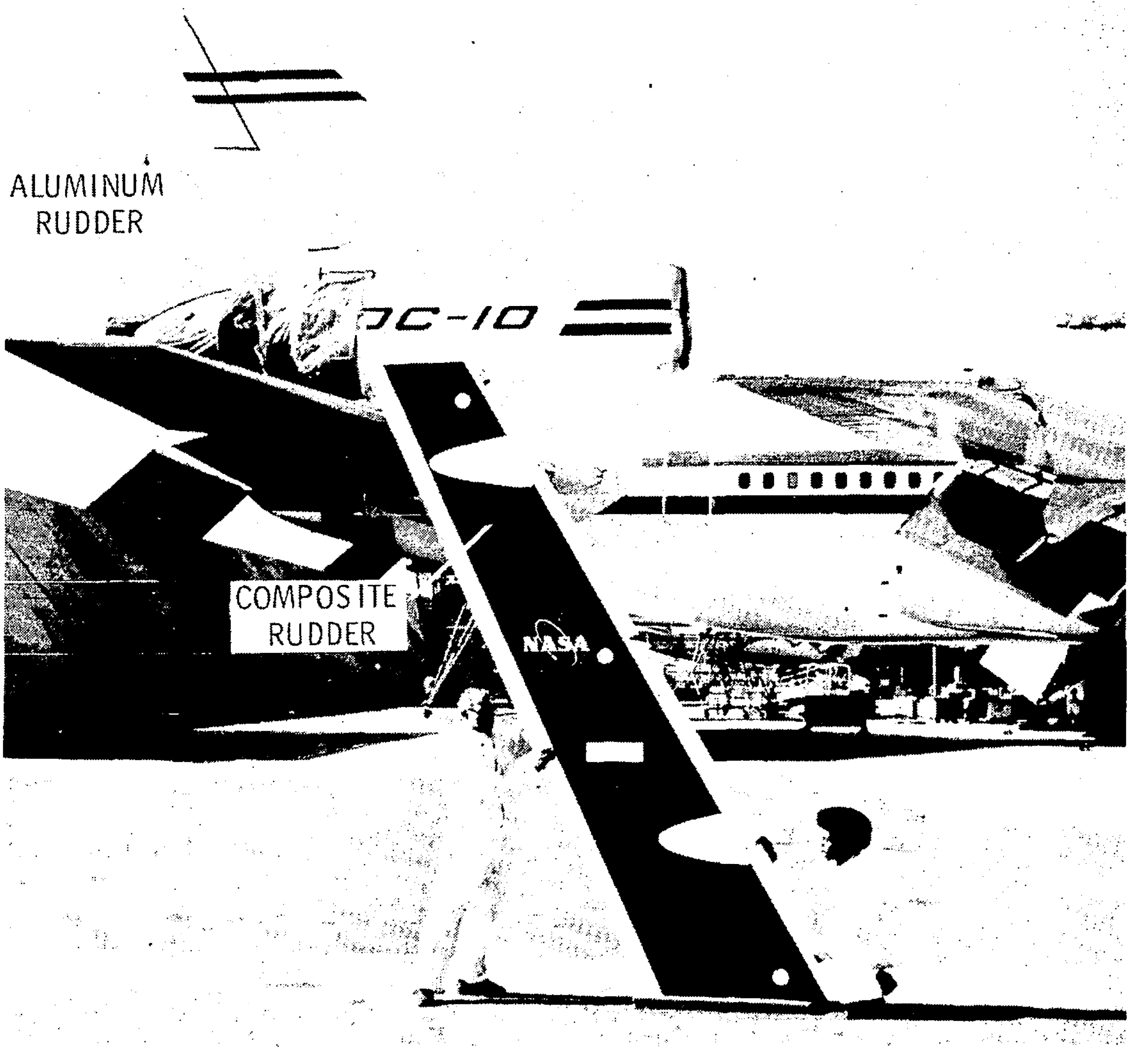
DOUGLAS DC-10 COMPOSITE RUDDER



LOCKHEED L-1011 COMPOSITE AILERON

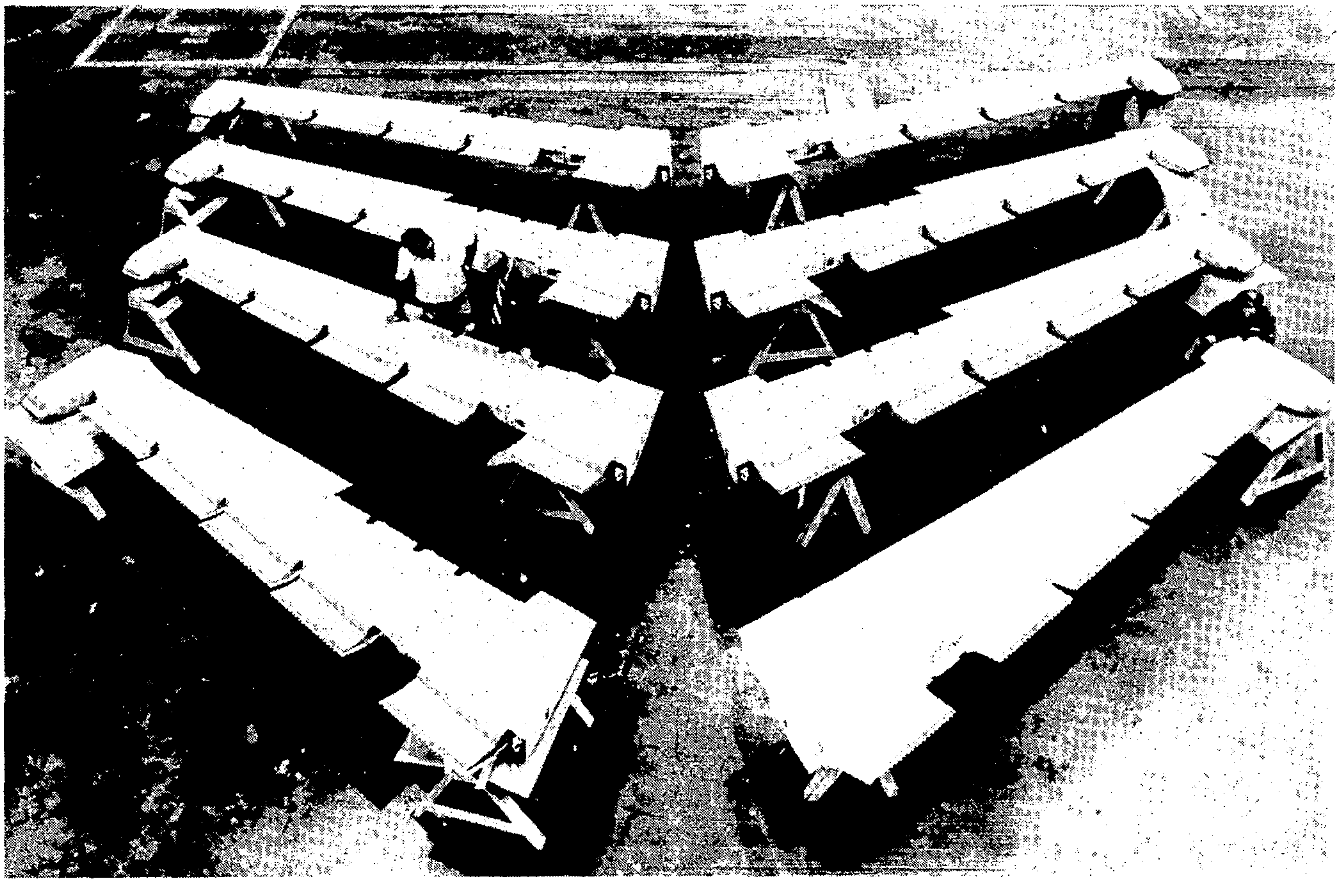
DC-10 GRAPHITE/EPOXY UPPER AFT RUDDER

The DC-10 composite rudder program is now complete and one of the rudders is shown here. The rudder is 3 feet at the root chord by 13 feet in length and weighs only 67 pounds, corresponding to a 26-percent weight savings over the metal design. Douglas has fabricated 20 rudders under this program and to date 13 have been placed in flight service with several domestic and foreign commercial airlines.



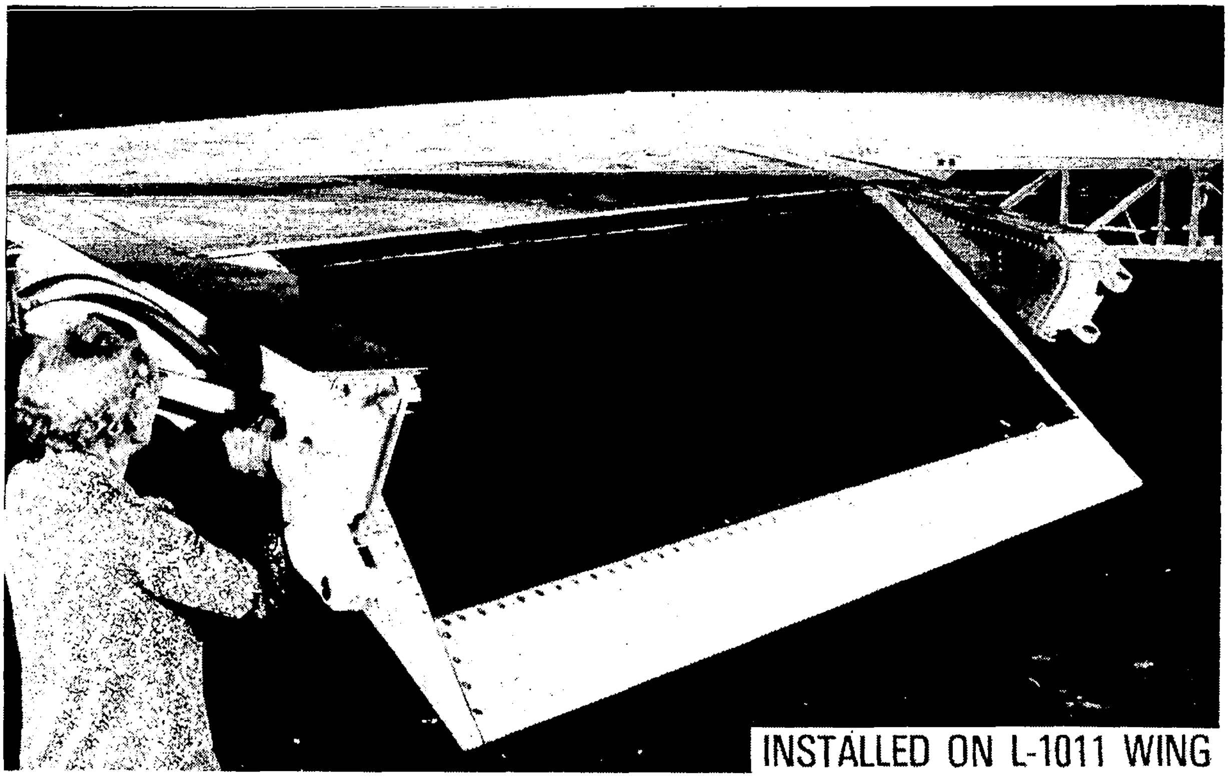
727 GRAPHITE/EPOXY ELEVATOR

In the Boeing 727 elevator program five shipsets of composite elevators have been fabricated and ground and flight tested. Four shipsets are shown here. This structure has been certificated by the FAA and all five shipsets have been placed into airline service with United Airlines. The elevators are 3-1/2 feet at the maximum chord by 17 feet in span and weigh 98 pounds. Weight reduction over the metal design is about 25 percent. The design and construction of these elevators provided the confidence and experience to place generically similar composite designs of secondary components on Boeing's new 767 and 757 aircraft.



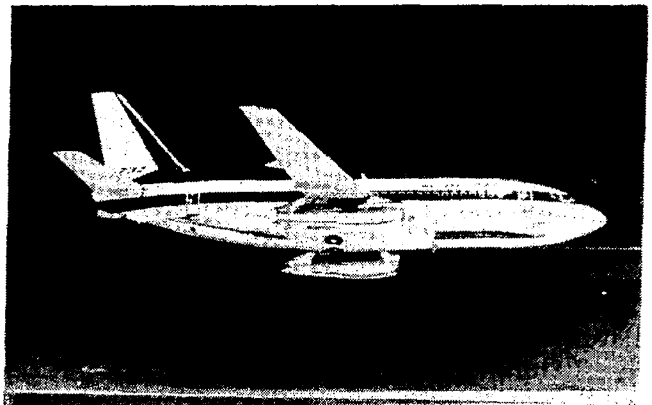
L-1011 GRAPHITE/EPOXY AILERON

The Lockheed L-1011 aileron program is also complete. Five shipsets of ailerons were successfully fabricated, ground and flight test qualified and certificated by FAA. One shipset is in service on the Lockheed Company airplane while the other four are in flight service on commercial aircraft, two with TWA and two with Delta. The ailerons are 4 feet by 8 feet and weigh 107 pounds, representing a 24-percent weight savings over the metal design.

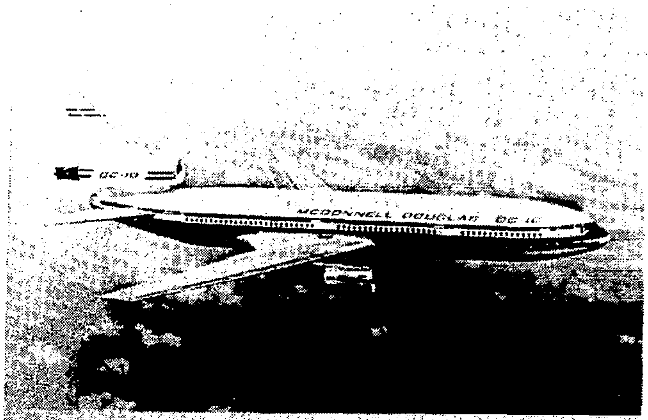

INSTALLED ON L-1011 WING

ACEE COMPOSITE MEDIUM PRIMARY STRUCTURES

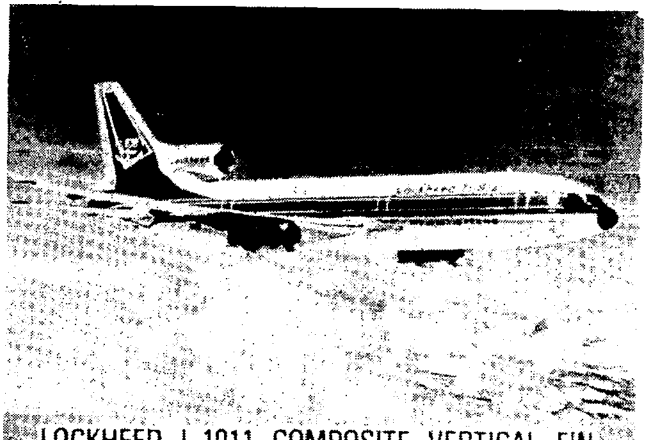
The commercial aircraft manufacturers are also developing composite versions of empennage primary structures on their existing aircraft. The empennage components offer a significant increase in challenge for composites application compared to the secondary structures. In particular, physical size, design requirements, load interaction, manufacturing and tooling each present formidable technology development tasks beyond those required for secondary structure. Components selected for redesign are the horizontal stabilizers on the Boeing 737 and the vertical stabilizers of the Douglas DC-10 and the Lockheed L-1011. These programs are still in progress.



BOEING 737 COMPOSITE HORIZONTAL STABILIZER



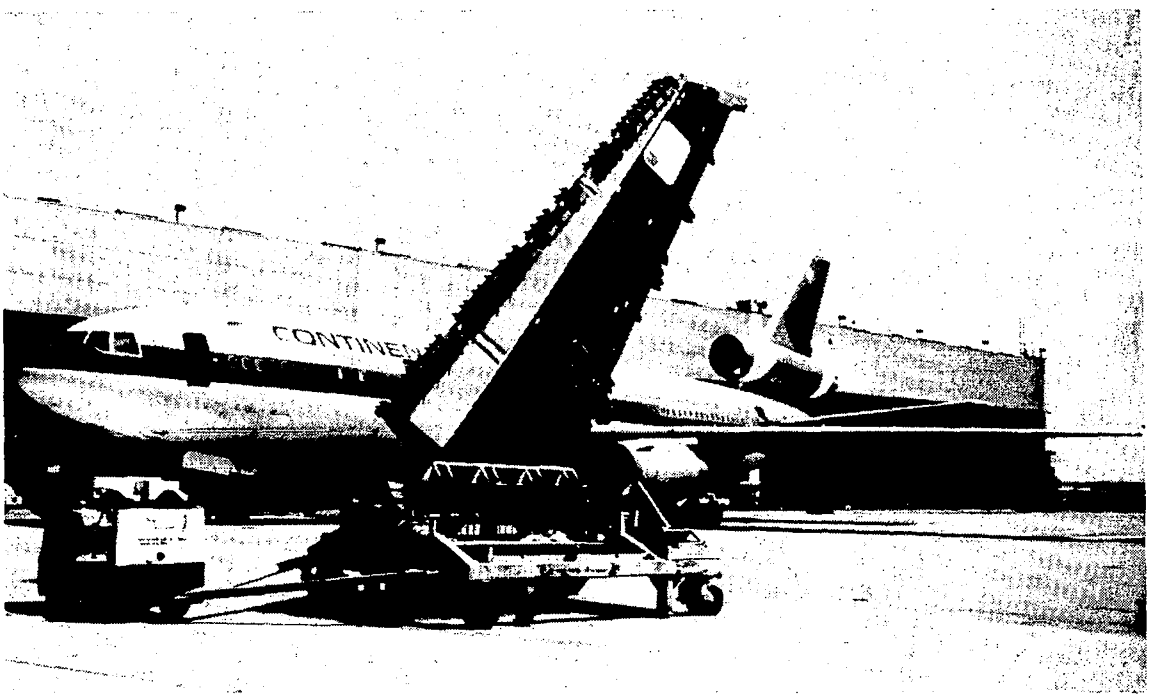
DOUGLAS DC-10 COMPOSITE VERTICAL STABILIZER



LOCKHEED L-1011 COMPOSITE VERTICAL FIN

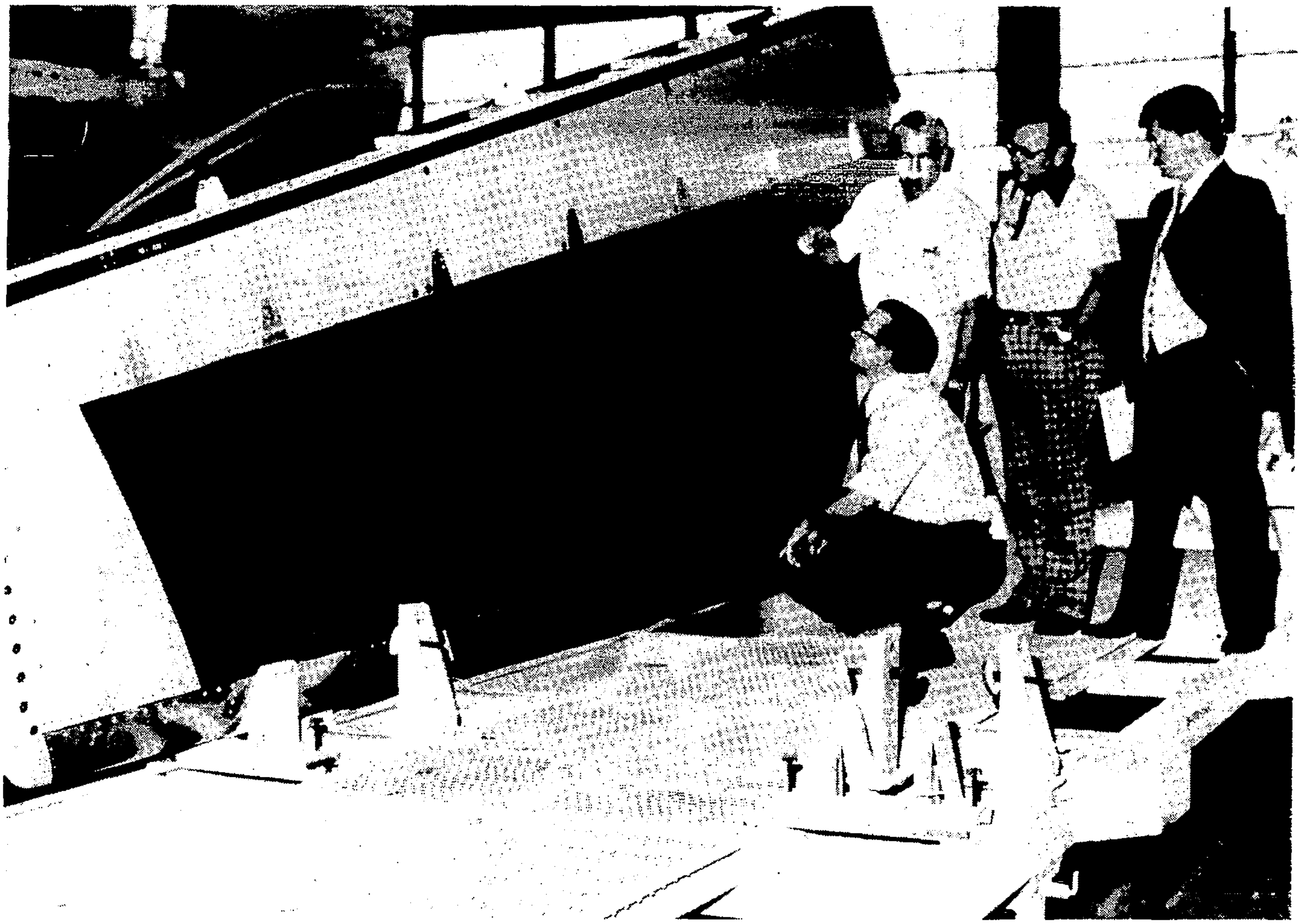
DC-10 GRAPHITE/EPOXY VERTICAL STABILIZER

All subcomponents (covers, spars, and ribs) of three full-scale composite vertical stabilizers for the DC-10 have been successfully fabricated and one stabilizer has been assembled for detailed ground qualification testing. The stabilizer is 7 feet at the root chord and 23 feet in span and weighs 779 pounds, representing a 23-percent weight savings over the metal design. The ground test program, which began in December 1981, was interrupted by premature failure initiating at the rear spar. Cause of failure has been identified and design modifications incorporated in the other two units. The second unit is being assembled to complete ground qualification testing, and the third unit will be flight tested to complete requirements for FAA certification. The third unit will then be placed in flight service with a commercial airline.



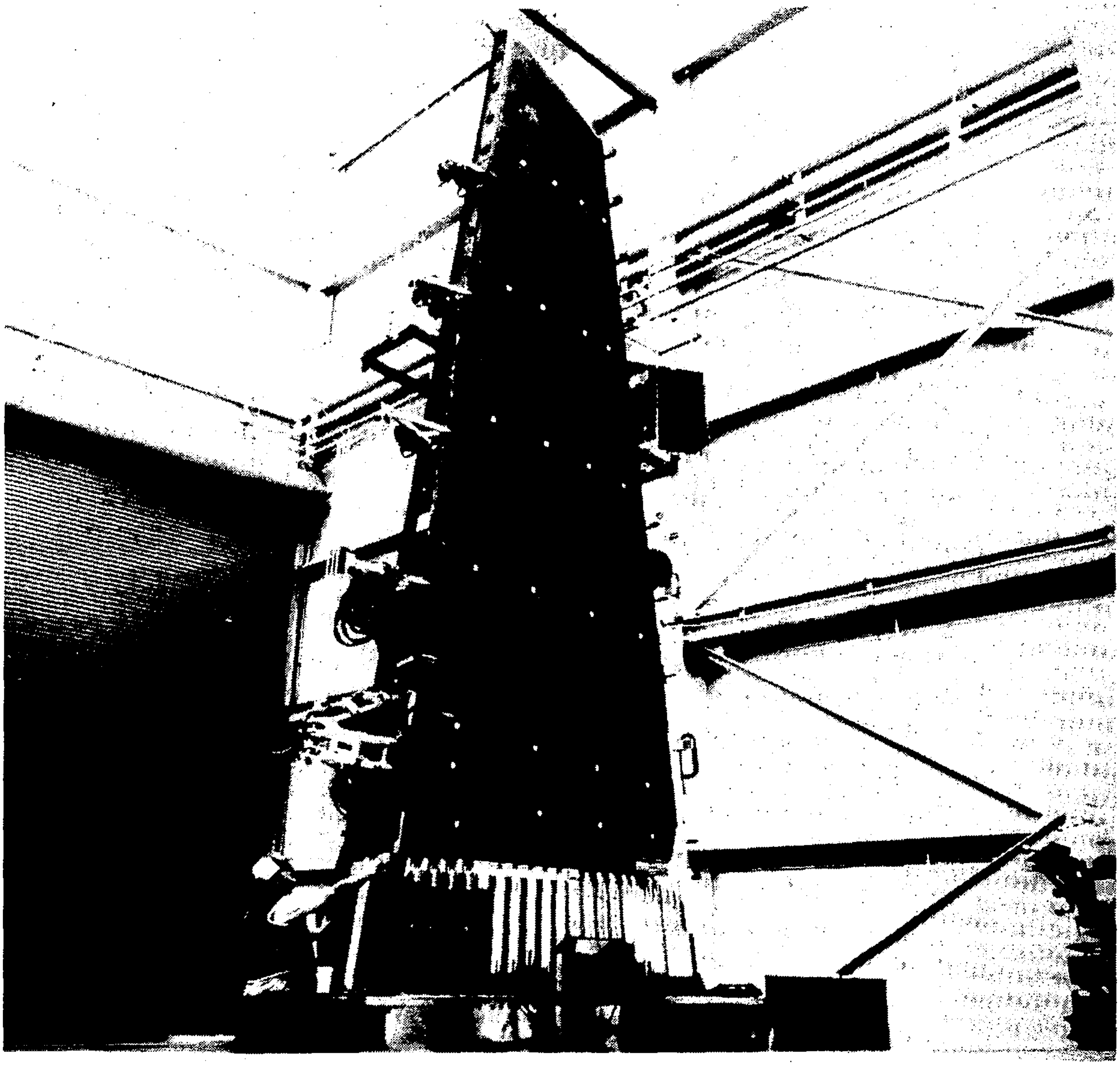
737 GRAPHITE/EPOXY HORIZONTAL STABILIZER

The 737 composite horizontal stabilizer is the smallest of the three empennage components and measures 4 feet at the root chord by 17 feet in span. The composite stabilizer weighs 204 pounds, which is a 22-percent weight reduction from the metal design. All ground and flight testing was completed successfully and FAA certification was granted in August 1982. This marks the first FAA certification of an empennage component for commercial transport aircraft. Boeing has fabricated and assembled five shipsets which are expected to be placed in flight service in the near future.



L-1011 GRAPHITE/EPOXY VERTICAL FIN

The L-1011 composite vertical fin is the largest in planform of the empennage components with a root chord of 9 feet and a span of 25 feet. The composite fin weighs 622 pounds, which is 28 percent lighter than the metal design. Two complete L-1011 composite fins have been assembled and one of these failed during ground qualification test just below design ultimate load, revealing a minor but significant design flaw. (Details of the failure will be discussed later.) The other fin was strengthened and successfully completed ground tests in June 1982 to qualify the modified design. Flight testing of the L-1011 composite fin is not planned.



ACEE COMPOSITE COMPONENT STATUS

Key features of the three secondary and empennage primary components are summarized in this table. Each component was manufactured in a production environment as a direct replacement of existing metal components. Weight savings from 22 to 28 percent were obtained even though these composite designs were driven to some degree by the existing metal design requirements. All components placed in flight service are tracked by the airlines and the manufacturer and are inspected periodically to insure safety of flight.

COMPONENT	SIZE ROOT X SPAN FT.	METAL DESIGN WEIGHT (LBS.)	COMPOSITE DESIGN WEIGHT (LBS.)	WEIGHT REDUCTION	NUMBER OF PRODUCTION UNITS	FAA CERTIFICATION	REMARKS
SECONDARY STRUCTURES							
DC-10 RUDDER	3.2 x 13.2	91	67	26.4%	20	MAY 1976	13 UNITS IN FLIGHT SERVICE
727 ELEVATOR	3.4 x 17.4	130	98	24.6%	11	JAN. 1980	5 SHIPSETS (10 UNITS) IN FLIGHT SERVICE
L-1011 AILERON	4.2 x 7.7	140	107	23.6%	12	SEPT. 1981	4 SHIPSETS (8 UNITS) IN FLIGHT SERVICE
MEDIUM PRIMARY STRUCTURES							
DC-10 V. STABILIZER	6.8 x 22.8	1005	779	22.6%	5	SEPT 1984	FLIGHT C/O AUG 1984
737 H. STABILIZER	4.3 x 16.7	262	204	22.1%	11	AUG. 1982	FLIGHT C/O COMPLETE
L-1011 V. FIN	8.9 x 25	858	622	28.4%	2	No	NO FLIGHT TEST PLANS

CHARACTERISTICS OF GRAPHITE/EPOXY MATERIAL

The remainder of this paper will highlight lessons learned in the ACEE composites program first by looking at attributes of the graphite/epoxy composite system and then by examining areas requiring advanced technology development.

Throughout the program the manufacturers required all fabrication and assembly of full-scale parts to be carried out in their production shops in order to obtain reliable manufacturing cost data as well as to gain valuable experience with production personnel. Hands-on experience was essential, not only in laying up and curing the laminate material with consistency and repeatability, but in developing test procedures to properly validate structural performance. This figure lists several positive characteristics of the composite system in manufacturing and performance areas that became evident during the full scale production phase of the program. The next few figures are used to illustrate these characteristics.

MANUFACTURING

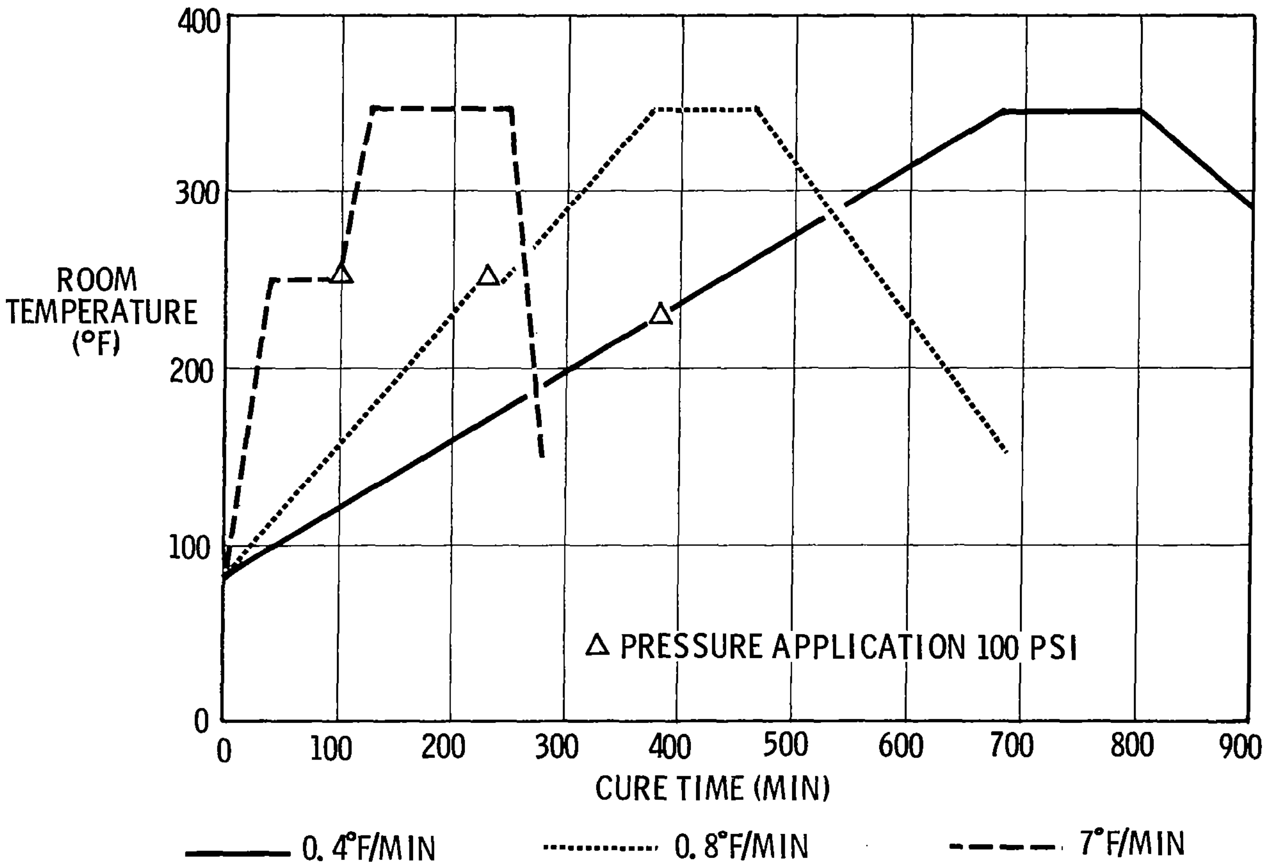
- TOLERANT RESIN CURE CYCLE
- UNIFORMITY OF HEAT DURING CURE OF PART IS ESSENTIAL
- PART SIZE LIMITED ONLY BY FACILITIES
- INNOVATIVE TOOLING PERMITS FABRICATION OF COMPLEX PARTS

PERFORMANCE

- FAILURE LOADS ARE PREDICTABLE
- FABRICATED PARTS ARE UNAFFECTED BY ENVIRONMENT

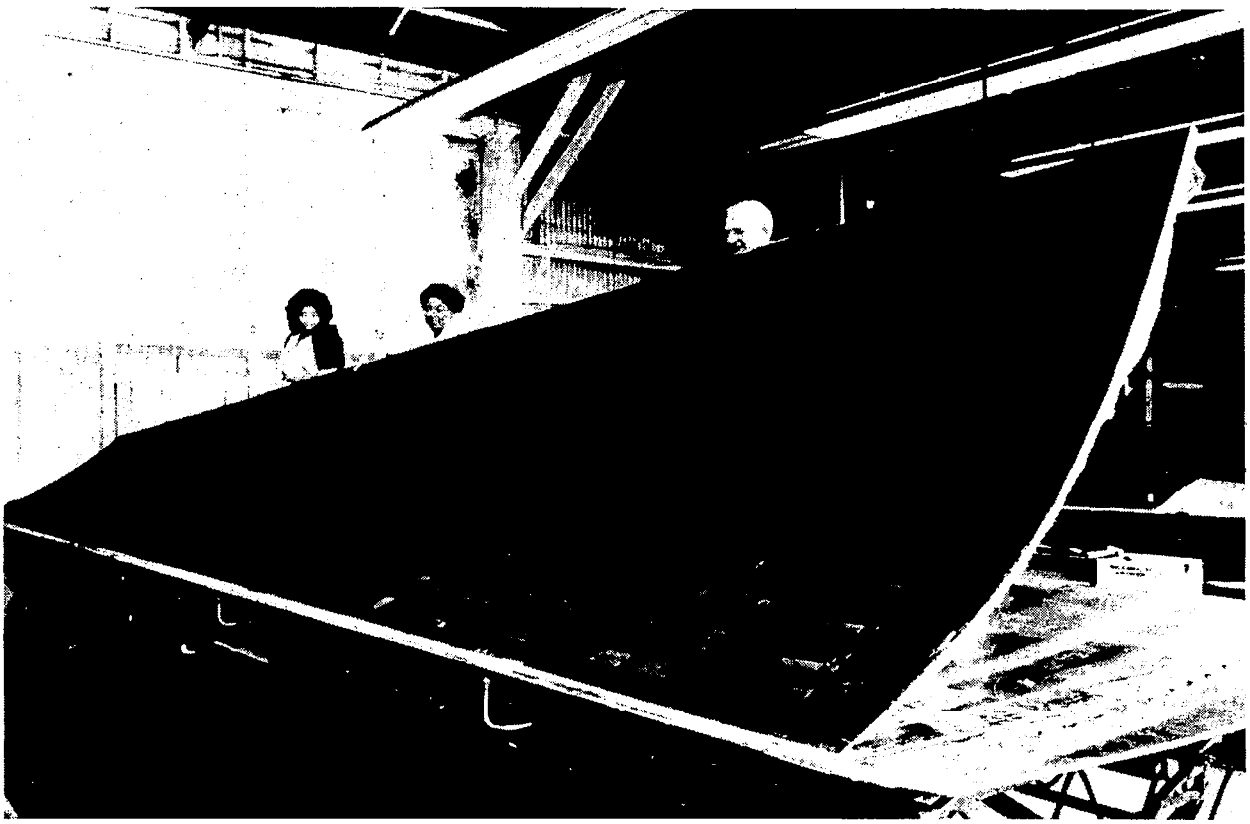
TYPICAL CURE CYCLES FOR GRAPHITE/EPOXY

The epoxy resin system used in the ACEE composites program is cured at 350°F and at high pressure (80 to 100 psi depending on the part). The structural performance of the part is found to be tolerant of the heat-up rate, a feature which offers considerable flexibility in utilization of facilities. A typical cure cycle matrix is shown in this figure. The three curves outline the fastest, average, and slowest heat rises used in an empennage program. The left and center curves show a heat rise to 250°F under vacuum, a dwell period, and pressure application. The right curve with a much slower heat rise has pressure applied at 225°F without a dwell period. The heat sink capacity of the tool generally determines the heat-up rate and care must be taken to assure uniformity of heat of the part during the cure cycle to avoid irregular resin flow.



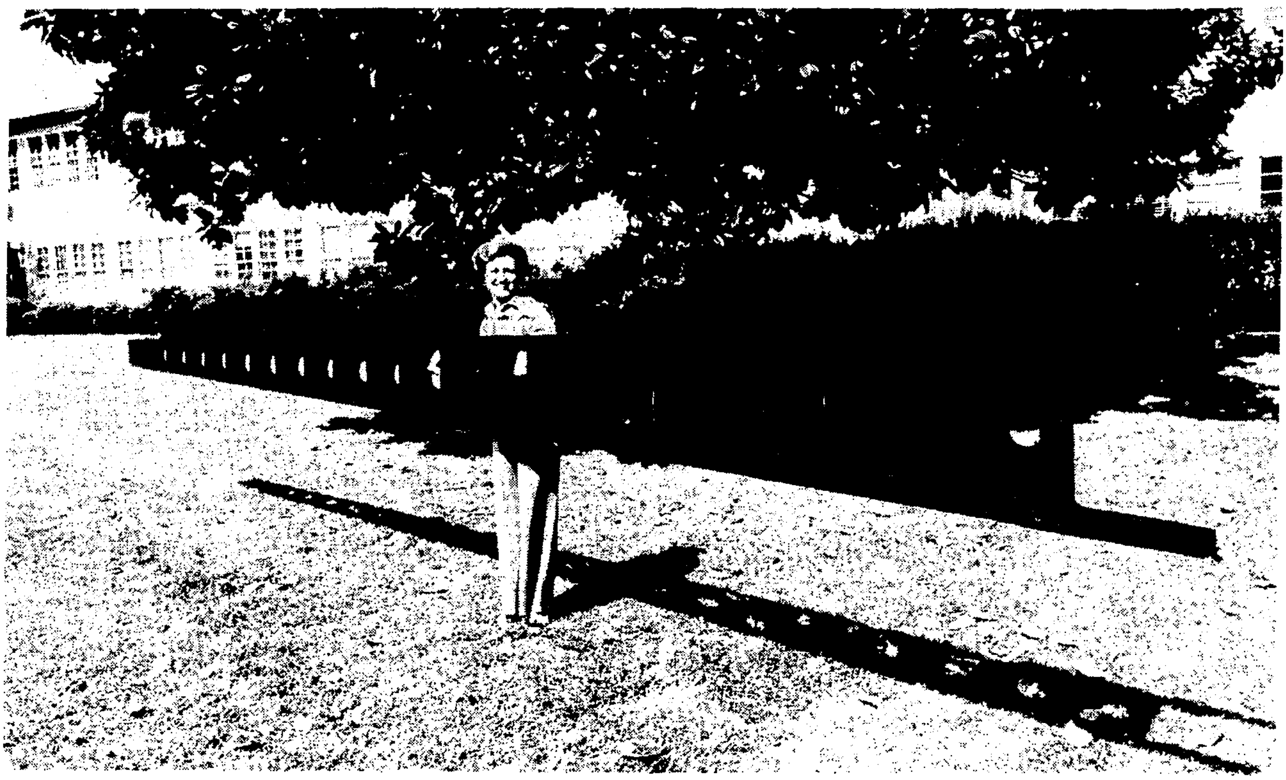
LARGE COMPLEX STRUCTURES MANUFACTURED

The largest single composite part in the empennage program is the cover panel of the L-1011 vertical fin. This structurally efficient hat-stiffened panel, which measures 8.9 feet at the root chord and 25 feet in span, was integrally cured in an autoclave and has no mechanical fasteners. The ease with which this part was cured suggests that the only limitation to part size may be the physical dimensions of the autoclave. The differential growth at cure temperature between the composite part and the hard tool (in this case steel) must, of course, be accounted for to assure dimensional control of critical elements and for tool release during cool-down following cure.



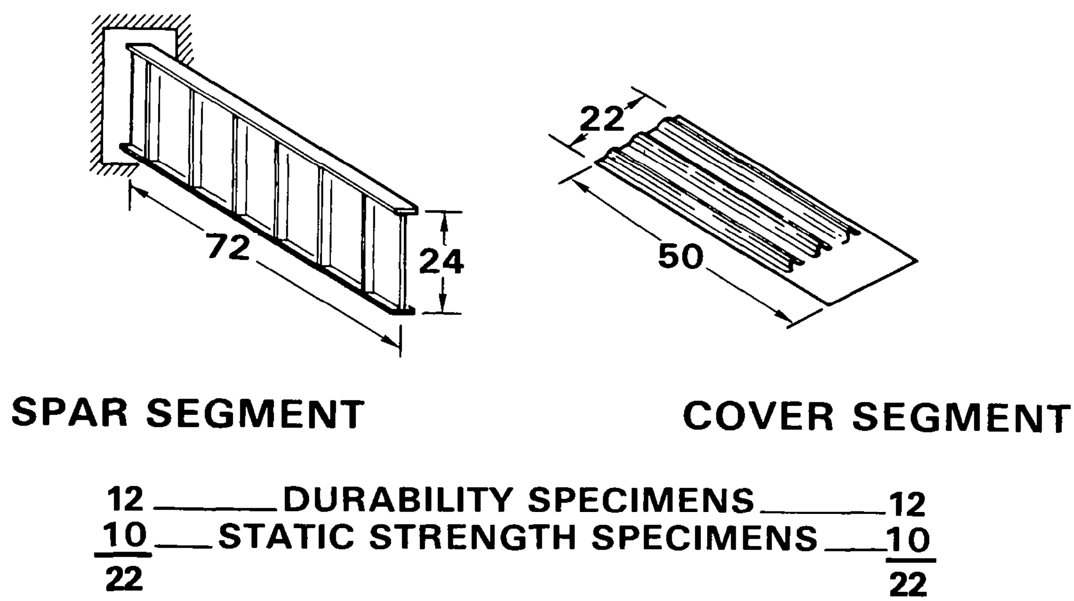
LIGHTWEIGHT STRUCTURAL COMPONENTS SIMPLIFY FINAL ASSEMBLY

One of the important features of the graphite/epoxy system is the unique opportunity to tailor stiffness properties and, with innovative tooling, to integrally mold efficient structural shapes that virtually eliminate mechanical fasteners. This figure illustrates one such example. The I-shaped spar configuration shown is 25 feet in length and is integrally (one step) cured, including the angle stiffeners on the shear web. Although some metallic parts are retained in final assembly, this single composite spar replaces 35 metal parts and 2,286 fasteners that were part of the all-metal design.



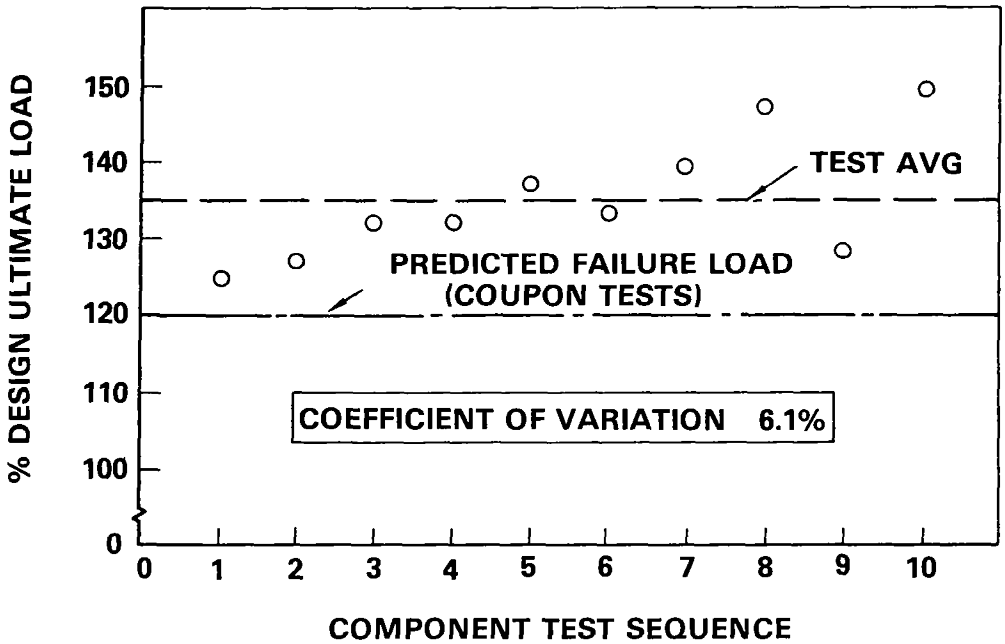
STRUCTURAL PERFORMANCE DEMONSTRATED

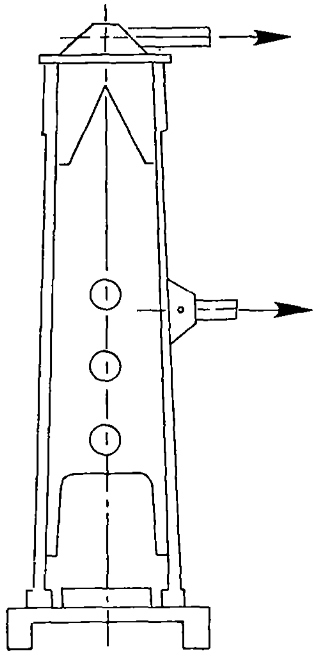
Following successful development of fabrication procedures for complex parts it is essential to validate the structural performance of cured parts. One program conducted by Lockheed to assess manufacturing tolerances on repeatability of performance is shown on the next three figures. Ten full-scale segments each of the L-1011 composite fin spar and cover were tested to destruction to obtain static strength characteristics. Each of the segments had measurable but acceptable manufacturing flaws such as thickness variations and small areas of porosity. The other twelve segments of each configuration are currently undergoing an accelerated 20 years of lifetime testing simulating flight cyclic loads, moisture, and temperature environments. Results from the static test program are shown in the next figure.



SPAR STATIC STRENGTH TESTS FOR MANUFACTURING VARIANCES

The results from the static strength tests of ten duplicate spar segments are shown in this figure. The spars were loaded in bending to produce a distribution of strain in the critical region equal to that of the full-size spar under flight loads. The load at failure is plotted as a percent of design ultimate load (DUL), which is 1.5 times the maximum aerodynamic load expected in flight. The test average is 135 percent DUL and about 15 percent higher than the predicted value based on conservative material properties from coupon tests. More importantly, all test values exceeded the predicted value, suggesting that the allowable manufacturing flaws have little effect on static strength.





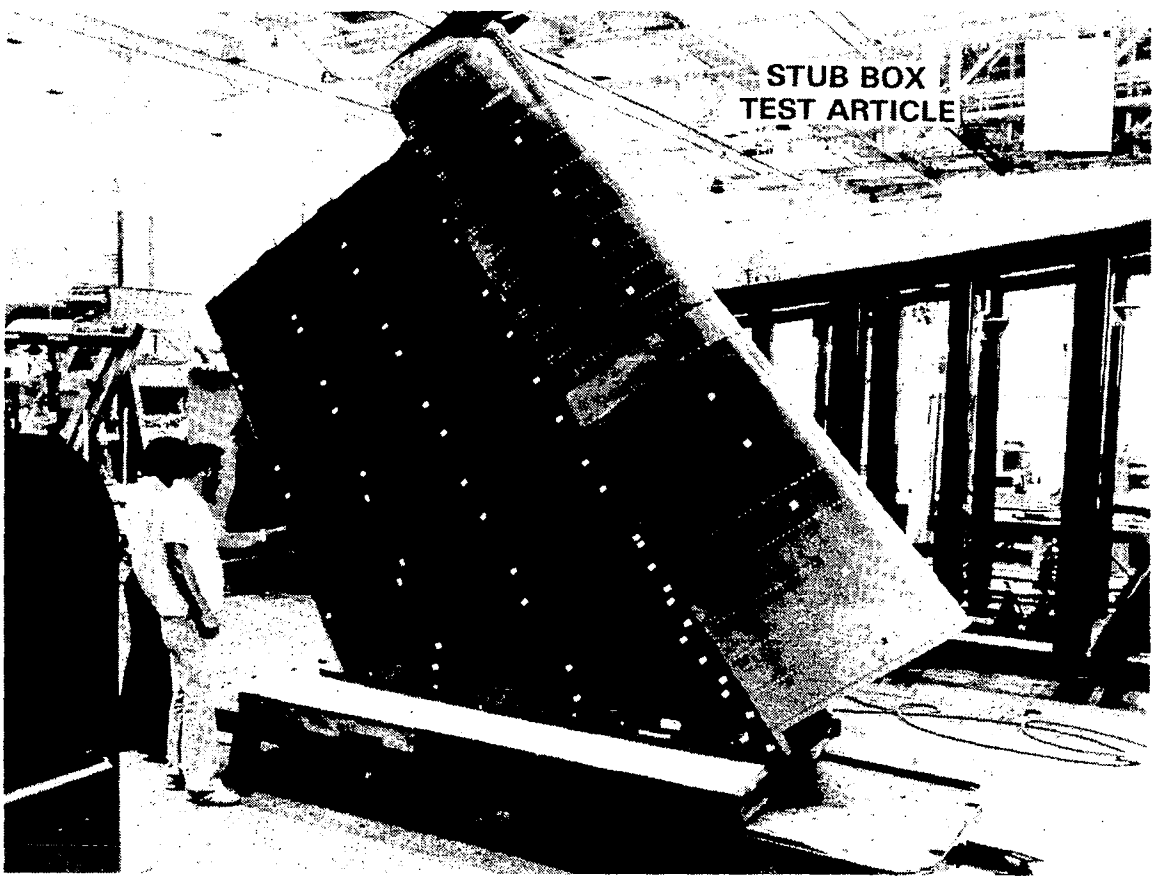
COMPARISON OF STATIC STRENGTH FOR
STRUCTURAL MATERIALS

The coefficients of variation of the ten cover and spar segments were 3.3 and 6.1, respectively. These values, obtained from tests of complex structural components, are consistent with values obtained from composite coupons and other structural materials and attest to the predictability and repeatability of performance of composite parts manufactured in a production surrounding.

MATERIAL	COMPONENT	SPEC. NO.	LOADING	COEFFICIENT OF VARIATION PERCENT
Graphite-Epoxy	PRVT-Cover	10	Compression	3.3
Graphite-Epoxy	PRVT-Spar	10	Bending	6.1
Graphite-Epoxy	Spoiler	15	Bending	6.6
Graphite-Epoxy	Laminate Coupons	411	Tension	5.7
Graphite-Epoxy	Laminate Coupons	411	Ten-Modulus	4.0
Graphite-Epoxy	Laminate Coupons	290	Compression	9.0
Graphite-Epoxy	Laminate Coupons	290	Compr-Modulus	5.2
Wood	Mosquito Wings	5	Bending	10.3
Wood	Plywood Shear Wall	27	Shear	9.7
Concrete	Test Cylinders	216	Compression	10.6
Aluminum	7049-T73 Die Forging	384	Tension	3.2
Aluminum	A357-T6 Casting	804	Tension	5.5
Titanium	TI-SAL-2.5SN Sheet	565	Tension	3.9
Steel	Structural Steel	3982	Tension	7.1
Steel	17-7PH Sheet	88	Tension	5.1

INFLUENCE OF ENVIRONMENTAL PARAMETERS ON STRUCTURAL PERFORMANCE

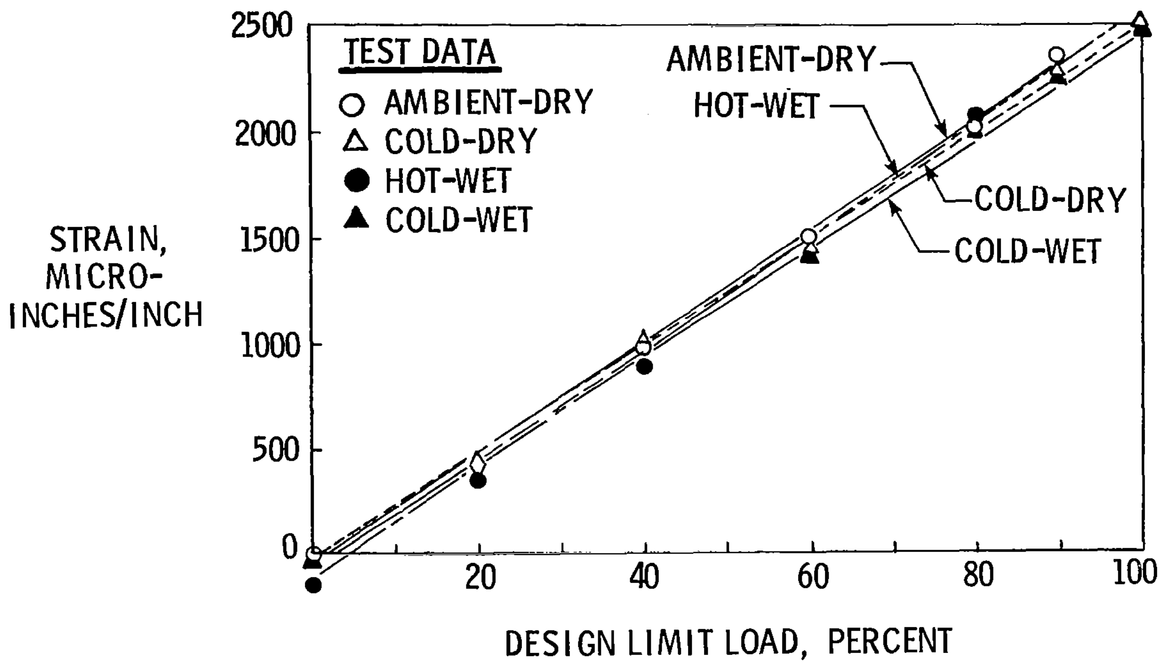
Another important issue that required full-scale components to evaluate was the influence of flight environmental parameters (moisture and temperature) on the performance of the structure under flight loads. A key activity designed to assess the influence of variations of flight temperature and moisture on strain distribution in a three-dimensional structure was carried out by Douglas on a stub box component shown in this figure. This component is the lower 45 percent of the DC-10 composite vertical stabilizer. The entire stub box, including the metallic leading edge, was mounted in an environmental chamber and subjected to a series of tests to determine, among other things, if the range of temperature and moisture to be expected in routine flight would affect the distribution of strain. The article was tested to limit load in bending, torsion, and shear at ambient conditions (72°F and no moisture (dry)), and then in bending at 0°F. This was followed by moisture conditioning at 170°F and 95 percent humidity for 14 days to achieve about 1 percent by weight moisture in the laminated structure. The component was then cyclic loaded for one lifetime at 0°F/wet and again tested to limit load in bending at 0°F and 130°F Following a second lifetime of cyclic loading and another limit load bending test at 0°F/wet the stub box was then loaded in bending to failure. Measured strains for the different test conditions are shown on the next figure.



INFLUENCE OF MOISTURE AND TEMPERATURE ON STRAIN

The stub box strains shown are taken from the same strain gage at a cutout in the web of the rear spar. Data are plotted as a function of limit load for the four environmental conditions - ambient/dry, cold/dry, hot/wet, and cold/wet. The strain variation between the test conditions is small and suggests a negligible effect of moisture and temperature on strains in the three-dimensional structure.

REAR SPAR CUTOUT STRAINS



KEY AREAS FOR TECHNOLOGY ADVANCEMENT

With the ACEE composites program now near completion it is interesting to examine the state of technology, especially in light of requirements of the future. While the current material system is clearly adequate for application to lightly loaded (secondary and empennage) structure, there are identifiable areas where technology advancement could significantly enhance the utilization of composites in primary wing and fuselage components. Three areas for technology advancement of particular importance to primary structure application are listed on this figure. Of these, secondary loads are probably the least understood. It is agreed among designers that there is a general state of uncertainty with composites as to the source, magnitude, and effects of secondary loads. Yet secondary loads are virtually impossible to eliminate from a complex built-up structure. While these loads are properly ignored in metallic structures, the sensitivity of current composite materials to interlamina forces can lead to serious weaknesses being designed into a composite structure. Such loads may be produced by eccentricities, irregular shapes, stiffness changes, and discontinuities, and their effects are magnified by the nonyielding nature of composites, which precludes load redistribution due to plasticity effects. The influence of secondary loads was vividly illustrated during ground tests of the L-1011 vertical fin, an event discussed on the two subsequent figures.

- **IMPROVED UNDERSTANDING OF SOURCE, MAGNITUDE, AND EFFECTS OF SECONDARY LOADS**

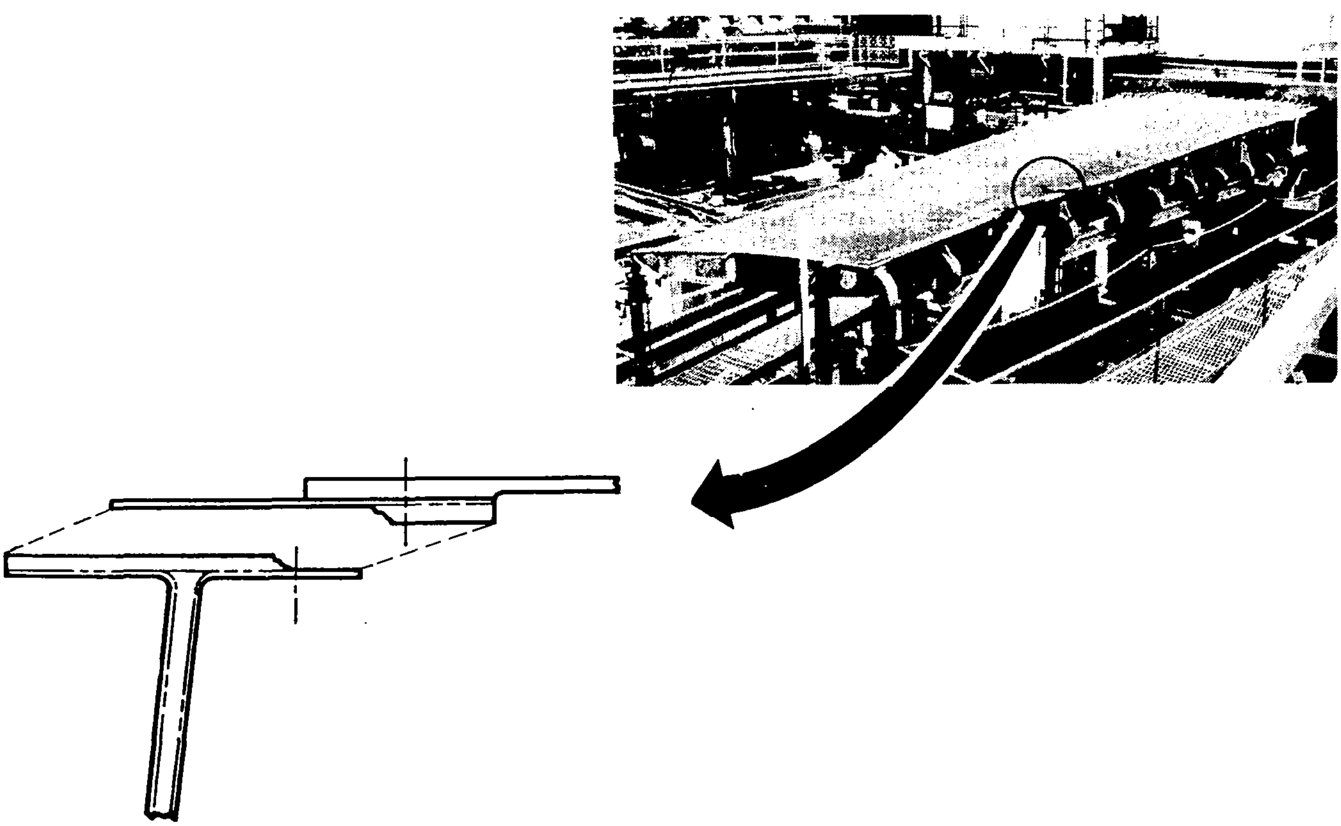
- **ADVANCED MANUFACTURING PROCESSES**

- **TOUGHER RESIN TO IMPROVE DAMAGE TOLERANCE AND DURABILITY**

EFFECTS OF SECONDARY LOADS IN COMPOSITES

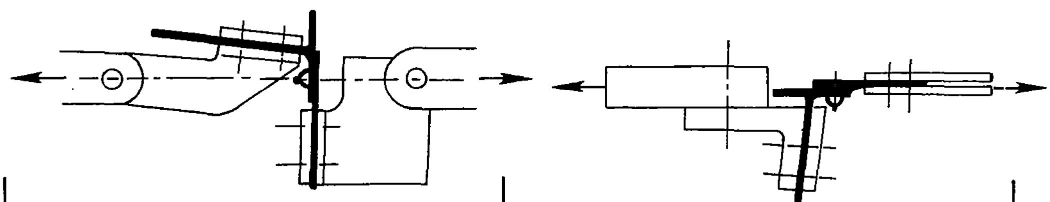
The L-1011 composite vertical fin failed during ground verification tests at 98 percent of design ultimate load (DUL) - 28 percent less than the predicted failure load. The failure caused separation of the cover and front spar, as shown in this figure, along the entire length of the component. After a careful investigation, the cause of failure was determined to be due to secondary loads caused by local buckling of the cover at the cover/spar interface. While local buckling beyond limit load was allowed in the design, the influence of these loads on the integrity of the structure was not expected. The interlamina tension forces caused delamination of the spar cap and ultimately separation along the line of the fasteners. Results from post-failure tests to assess the influence of such loads are shown on the next figure.

TYPICAL SPAR CAP DAMAGE IN PRIMARY FAILURE ZONE



INFLUENCE OF LOAD CYCLING ON INTERLAMINA STRENGTH

Tests were conducted to determine the strength of the designs of the spar/cover intersection for secondary loads causing interlamina tension and transverse tension. The first test was on virgin material that had no prior loading and results were reasonable and indicated adequate margin. The estimated maximum interlamina tension load expected in flight was 68 pounds per fastener. The second test, on an undamage segment of the failed spar, showed large reductions in strength. This apparent influence of load cycling was verified by a third test on specimens subjected to load cycles similar to those of the ground test article prior to failure. This degradation in strength is the result of a design weakness in the spar cap, which can be avoided in future designs when criteria for secondary loads become a routine part of design practice.

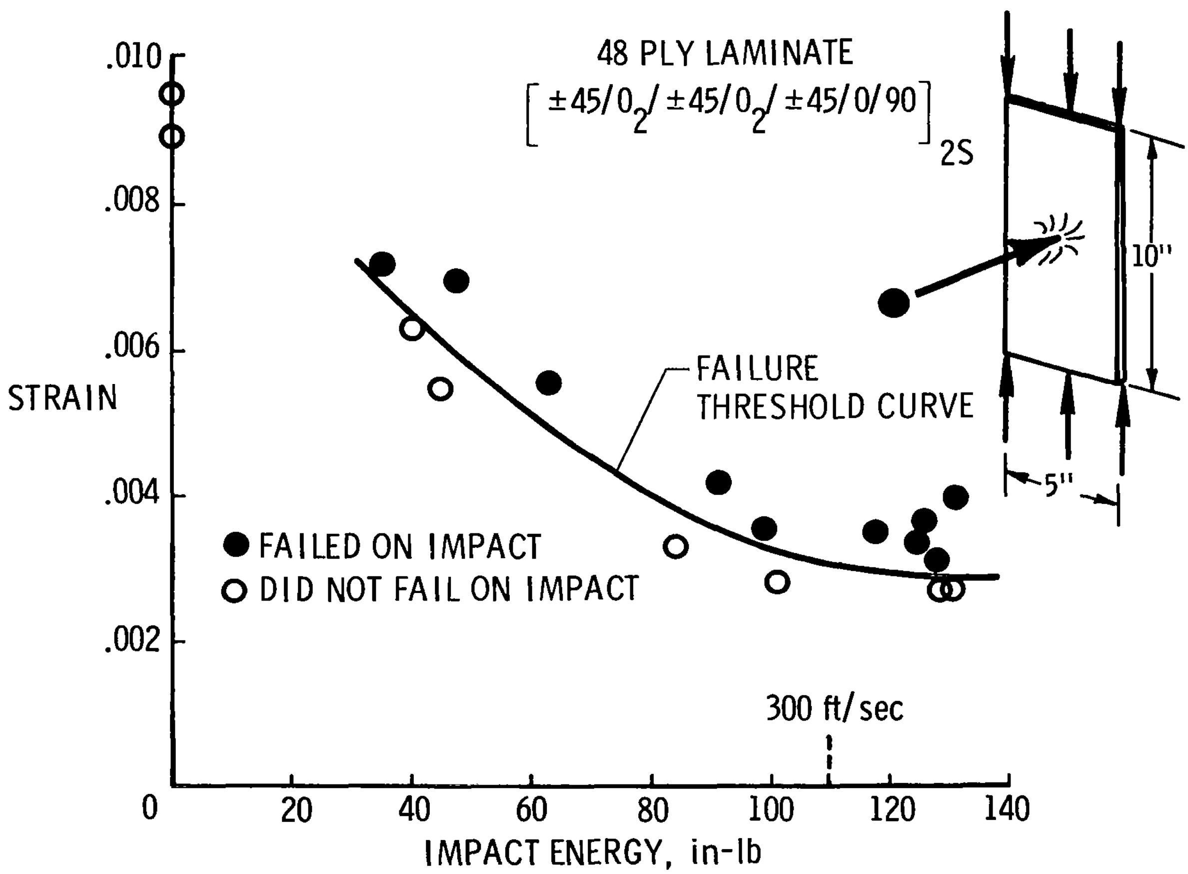


SPECIMEN CONDITIONING	INTERLAMINA TENSION (LB./FASTENER)	TRANSVERSE TENSION (LB./IN.)
NO PRIOR LOADING	88	445
SEGMENT OF SPAR OF FAILED GROUND TEST UNIT	34	256
PRIOR LOADING EQUAL TO GROUND TEST UNIT	56	225

INFLUENCE OF DAMAGE ON ALLOWABLE STRAIN

Another area in which technology advancement should pay off is the improvement of strain capability of the cured laminate. Limiting strain of current graphite/epoxy systems is determined primarily by sensitivity to impact damage. Typical results on this figure show the failure threshold of a compression-loaded graphite/epoxy plate as a function of impact energy. Designs of secondary and empennage structures have been governed primarily by stiffness requirements with design ultimate strains of about 0.003 μin./in.; and indeed, current resin systems can meet the strain requirements without a weight penalty. However, for large primary wing and fuselage structures, which are designed by strength requirements, higher strain capability is required if maximum utilization of composites is to be achieved. This is illustrated on the next figure.

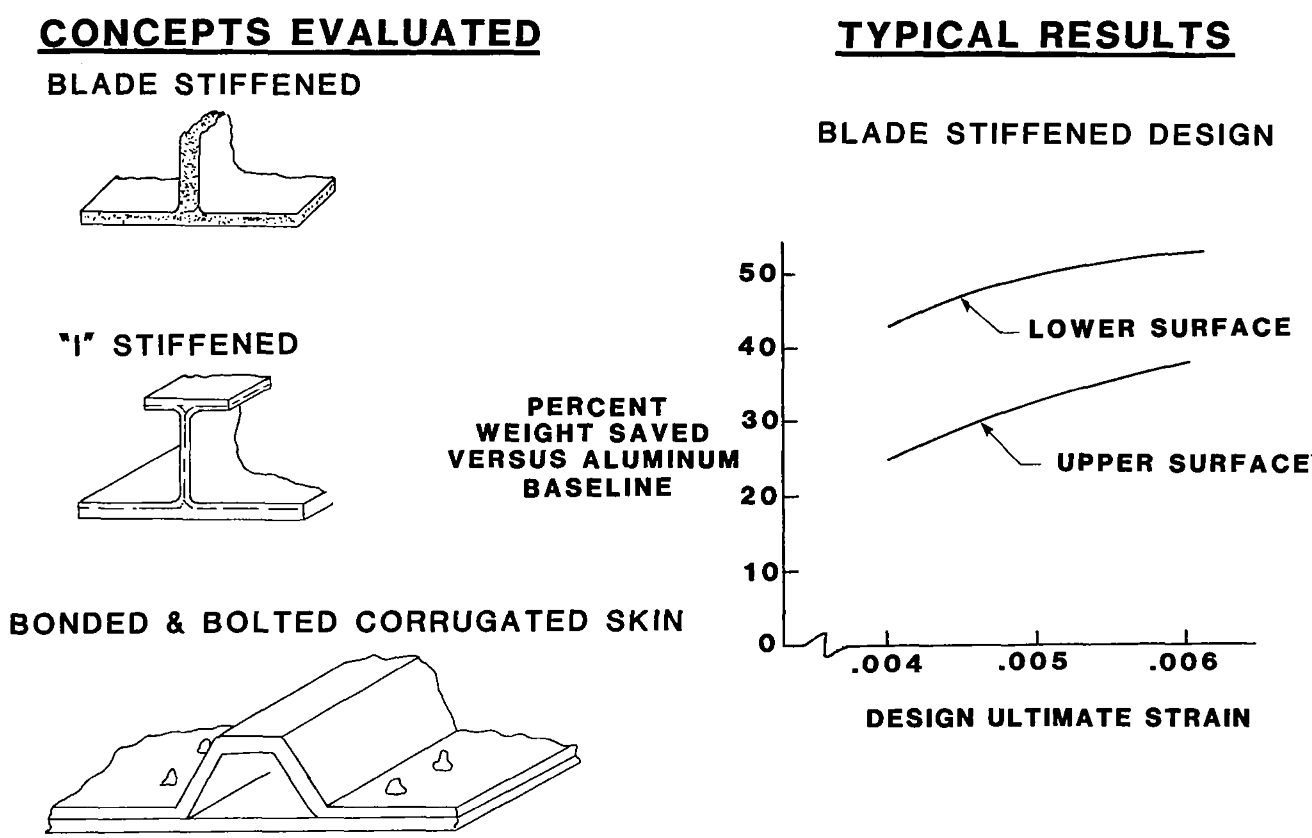
EFFECT OF PROJECTILE IMPACT ON COMPRESSION STRENGTH



WEIGHT EFFECIENCY OF PRIMARY STRUCTURE DEPENDENT ON ALLOWABLE STRAIN

Wing structure of large transport aircraft carry considerably higher unit loads than other parts of the vehicle and element designs require thick laminates that are inherently stiff; consequently, strength rather than stiffness is the primary design driver. Recent wing surface design studies for commercial transports indicate that significant weight savings are permissible if design strains are not limited by impact requirements. This figure shows three structural configurations for wing surface panel designs; the curves on the figure show weight savings as a function of design strain for an optimized composite "blade" stiffened design over an aluminium design for the upper and lower surfaces. As can be seen a significant increase in weight savings for wing surface panels is possible if design strain can be increased from 0.004 to 0.006 μin./in. NASA is actively pursuing research in constituent relationships within the matrix and fiber/matrix interaction to achieve the higher strain capability.

WING SURFACE DESIGNS



ACEE COMPOSITES PROGRAM CONCLUSIONS

The composite element of the NASA Aircraft Energy Efficiency (ACEE) program was initiated for the development of technology within the commercial airframe industry to foster the application of composites in future commercial aircraft. The program, which has been ongoing for nearly 6 years, has demonstrated the high potential for composites in commercial aircraft through the design and development of secondary and empennage primary components for existing aircraft. The composite material system used has been shown to be adaptable to variations in manufacturing processes, and it is readily formable into complex and highly efficient structural shapes with relatively good repeatability in performance. Large complex box structures have been fabricated, assembled, and ground tested and have provided a focus for technology advancements to further improve performance and assure flight safety.

The composites program has been very effective in developing confidence and experience within commercial airframe companies as engineering and manufacturing personnel have accepted and met challenges to develop and demonstrate weight and cost effective composite components. This level of confidence is strengthened by the generally widespread application of secondary composite components in flight service stemming from the ACEE composites program or prior NASA programs. This flight service experience is providing much needed airline participation and should pave the way for more committed involvement in composites in empennage primary and large primary structures in the next generation of aircraft.

- **LARGE STRUCTURAL COMPONENTS HAVE BEEN SUCCESSFULLY DESIGNED, FABRICATED, AND VERIFICATION TESTED**
- **WEIGHT SAVINGS UP TO 27 PERCENT DEMONSTRATED**
- **DESIGN AND MANUFACTURING CONFIDENCE IS HIGH**
- **TECHNOLOGY IS READY FOR SELECTIVE APPLICATION OF COMPOSITES IN NEW AIRCRAFT**
- **COMPONENTS PROGRAM PROVIDED FOCUS FOR ADVANCED RESEARCH**