NASA Technical Memorandum 84627

Ground Test Experience With Large Composite Structures For Commercial Transports

H.L. Bohon, A.J. Chapman, III H.A. Leybold

MARCH 1983

NASA
National Aeronautics and
Space Administration
Langley Research Center
Hampton, Virginia 23665

GROUND TEST EXPERIENCE WITH LARGE COMPOSITE STRUCTURES FOR COMMERCIAL TRANSPORTS

Herman L. Bohon, Andrew J. Chapman, III, and Herbert A. Leybold
NASA Langley Research Center
Hampton, Virginia

Abstract

Composite technology applicable to transport empennage structures has been developed through contracts sponsored by the NASA Aircraft Energy Efficiency (ACEE) Program Office. The empennage components, the horizontal stabilizer for the Boeing 737, the vertical fin for the Lockheed L-1011, and the vertical stabilizer for the Douglas DC-10, have been designed to replace the existing metal structures without modification of other parts of the aircraft. A principal element in the program for each component is an extensive ground test series of a full size structure. The programs for two of the components, the 737 Horizontal Stabilizer and the DC-10 Vertical Stabilizer, include, as objectives, FAA certification and airline flight service. The ground test series for these components are, consequently, essential steps in verifying compliance with FAA certification requirements. The initial ground test of each component resulted in structural failure at less than ultimate design loads. While such failures represent major program delays, the investigation and analysis of each failure revealed significant lessons for effective utilization of composites in primary structure. Foremost among these are secondary loads that produce through-the-thickness forces which may lead to serious weaknesses in an otherwise sound structural design. The sources, magnitude, and effects of secondary loads need to be thoroughly understood and accounted for by the designers of composite primary aircraft structures.

Introduction

In recent years, graphite/epoxy composite material has had widespread application in military aircraft and to a more limited degree in components of 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 have significantly better performance than current aircraft. Studies have shown that the use of composite materials for transport aircraft structures provides the opportunity to reduce structural weight by as much as 25 percent over current aluminum structures with a corresponding reduction in fuel consumption of 12 to 15 percent (see references 1, 2, and 3).

In order to establish a basis to assess the potential benefits of composites, the NASA Aircraft Energy Efficiency (ACEE) Composites Office sponsored technology development programs with Boeing Commercial Airplane Company, Lockheed-California Company, and Douglas Aircraft Company. The primary objective of the ACEE program was to develop the essential technologies to permit the efficient utilization of composites in airframe structures of future transport aircraft. The transport manufacturers were challenged to redesign selected secondary and medium-primary components on existing aircraft with composite material and to validate the weight and cost benefits and structural efficiency of these components through serial fabrication and detailed ground tests.

The secondary component programs have now been completed (references 4, 5, and 6) and several components are in flight service on aircraft on domestic and foreign commercial airlines. The medium-primary component programs have been carried through a series of qualification ground tests. The initial ground test of each component resulted in structural failure at less than ultimate design load. Subsequent investigation and analysis of each failure revealed significant lessons for effective utilization of composites in large transport structures. Ground test experience from these medium-primary component programs will be reviewed in this paper.

Transport Medium-Primary Component Program

Contracts for the development of composites technology applicable to empennage structure of transport aircraft were initiated in 1977 with the three major airframe manufacturers. The objective of these contracts and components selected for development are shown in figure 1. The components include the vertical fin of the L-1011 aircraft, the vertical stabilizer of the DC-10, and the horizontal stabilizer of the 737 aircraft. Significant weight savings were achieved for the actual composite components shown in the photographs. These weights ranged from 22 to 28 percent less than the comparable aluminum component.

The objective of the medium-primary program is to provide the opportunity for the transport manufacturers to obtain the technology and gain the confidence required for a commitment to production of composite structures of generically similar construction. To achieve this objective the manufacturers must develop not only know-how for low-cost fabrication and designs with predictable performance, but enough test and actual manufacturing experience to accurately predict durability for structural warranty purposes and costs for product pricing. The program must also demonstrate flightworthiness for certification by the FAA and maintainability for acceptance by the airlines.

The major elements of the technology development program are identified in figure 2. Various material options were evaluated before selecting one and then extensive testing was conducted to develop an adequate data base of material design properties. The material selected by the three manufacturers was the Thornel 300 fiber with Narmco 5208 resin. Numerous design options for major subcomponents (covers, spars, and ribs) of the empennage structures were evaluated on the basis of weight efficiency, fabricability, maintainability and inspectability, and design options were narrowed through analysis and a varied spectrum of development tests on small and large elements. The program also included the development of a suitable production process including economical laminate preparation and the appropriate combination of temperature and pressure during cure of the structural parts. Tools were designed and fabricated, and full-scale components were then manufactured for ground qualification tests, flight tests, and airline service. Though serial production was limited in this program, fabrication included five shipsets of 737 horizontal stabilizers and three units each of the L-1011 vertical fin and the DC-10 vertical stabilizer. Most of the element, subcomponent and component test data and associated analyses are included in a report submitted to FAA for flight certification, which must precede airline service. Inspection and repair methods to insure adequate maintenance in service were also part of the development program. Principal results from the ground qualification tests on the three empennage structures are the focus of this paper.

Component Designs

737 Composite Horizontal Stabilizer.- The structural configuration of the 737 composite horizontal stabilizer is shown in figure 3. This component is the smallest of the three medium-primary structures and measures 4 feet at the root chord by 17 feet in span. The covers are I-stiffened with the stiffeners and skin integrally cured, whereas the front and rear spars are precured channel sections secondarily bonded back-to-back with web stiffeners mechanically attached. The ribs are also channel sections and have honeycomb stabilized webs. All components are assembled with mechanical fasteners. Load transfer from the stabilizer to the fuselage carry-through structure is by thick graphite lugs with metal face-plates, two on the front spar and three on the rear spar. Concentrated loads from the lugs are dispersed into the spar web by thick precured chord elements indicated by section A-A in figure 3. Component design and manufacturing details are reported in references 7 through 11.

L-1011 Composite Vertical Fin.- The structural arrangement of the fin is shown in figure 4. The vertical fin is the largest in planform of the three empennage structures and measures approximately 9 feet at the root by 25 feet in span. All subcomponent parts of the structural box are fabricated with graphite-epoxy except the aluminum truss members of seven truss ribs. The design features integrally cured hat-stiffened covers and stiffened web spars. The integrally formed composite I-shaped spars replace 35 metal parts and over 2200 fasteners required in the metal design. The covers, spars, and ribs are mechanically joined during final assembly. Component design and manufacturing details are reported in references 12 through 16.

DC-10 Composite Vertical Stabilizer.- The structural configuration of the stabilizer is shown in figure 5. Since aerodynamic loads from the stabilizer are transmitted to the airframe structure at four discrete spar locations in the metal design, retention of the multi-spar concept was necessary. Thus, in order to achieve a significant weight reduction, the composite design incorporated sine wave webs for all spars and ribs, and honeycomb sandwich covers. The covers consist of a grid of thick laminates which mate with and become an extension of the spar and rib caps and Nomex honeycomb core sandwich skins between spar and rib caps. Each of the subcomponents are integrally cured although the three longest spars are cured in two parts and spliced at a spanwise station. The spars and ribs are assembled as shown in the figure by secondary bonding along the webs of the spar/rib interfaces without the use of mechanical fasteners. The covers, when mechanically attached to the spar and rib caps, complete the assembly. Additional details on the DC-10 composite stabilizer are given in reference 17.

Qualification Tests

Ground Test Load Introduction.- Methods of applying simulated aerodynamic loads to the full-scale components during ground tests were different for each structure. These methods are shown schematically in figure 6. Static loads to the DC-10 vertical and the 737 horizontal stabilizers were applied to the covers through a whiffletree arrangement. Tension "whiffling" was used with the 737 horizontal stabilizer while compression "whiffling" was used with the DC-10 vertical stabilizer. In each case, loads were transmitted into the covers at pads bonded to the surface. Compression pads on the DC-10 component were restricted to locations over the spars to avoid concentrated loads on the honeycomb areas of the covers, whereas pads on the 737 stabilizer were placed at random to provide the best distribution of internal loads. Primary loads to the L-1011 fin were applied along the front spar web through a yoke device shown by the insert in figure 6. Simulated rudder kick loads on both vertical fin components were imposed by load jacks attached directly to the rudder hinges. The 737 horizontal stabilizer was tested with a production elevator in place which was loaded through a whiffletree in the same fashion as the main structural box.

Special care was taken to attach the components to test fixtures which would simulate as nearly as possible the reaction loads and stiffness characteristics of attachments on the aircraft. This was readily accomplished with the

737 horizontal stabilizer where a production center carry-through section was employed. The stabilizer was mated with the carry-through structure at the front spar with two lug pins and at the rear spar with three lug pins (see figure 3). A dummy stabilizer was loaded simultaneously to produce duplicate reaction loads at opposite lugs. Aerodynamic loads on the DC-10 vertical stabilizer are reacted at the four spars which, on the aircraft, are attached to "banjo frames" that house the aft engine. In the ground test setup, the stiffness properties of the banjo frames are simulated by aluminum tubes carefully designed to duplicate reaction loads and deflections at the four spar attachments. Loads from the L-1011 fin are carried into the airframe at both front and rear spars and along the cover and in order to simulate this reaction detailed transition structure was required (see figure 6). The aluminum transition structure was very stiff in the spanwise direction, but had sufficient flexibility in the chordwise direction to avoid stress concentrations at the graphite/aluminum interface.

<u>737 Composite Horizontal Stabilizer</u>.- The ground test article was subjected to a series of static and fatigue tests at ambient conditions to assess performance of the all-composite stabilizer under several flight conditions for direct comparison with calculated results. Initially, design limit load was applied representing three different critical flight conditions (i.e., shear, torsion, and bending) followed by one-half lifetime of spectrum fatigue loads. The article was then subjected to damage tolerance tests following visible surface damage inflicted on the covers and spars. These tests included one lifetime of spectrum fatigue followed by tests in bending to design ultimate load (150 percent of limit load). After damage tolerance testing, several fail-safe tests were conducted where the stabilizer was loaded to limit load with lug pin removed at the spar attachment to the carry-through structure. The stabilizer failed on the fourth and last of these tests at 91 percent of limit load.

The fail-safe test configuration of the rear spar lug when failure occurred is shown in figure 7. The upper lug pin was removed from the rear spar and the stabilizer was loaded in bending. The arrows depict the direction of reaction loads at the two pinned lugs. Failure was initiated in the web between the lug chords and propagated along the span. However, damage was constrained within the web of the rear spar. A photograph of the damaged spar is shown in figure 8 and the extent of crack propagation on both interior and exterior webs and at the web midplane is shown in figure 9. Two modes of failure are shown by the photographs in figure 10. The tension failure of the web, shown by Section B, is at the region of highest strain. Farther outboard from the lugs, at Section A, the failure was a delamination of the web plies that wrap around the lug chord. Strain measurements at two locations on the rear spar web are shown in figure 11. These data confirm the high diagonal tension strain near the point of failure initiation and indicate nonlinear behavior at this location after about 50 percent of limit load.

Post-test analysis with a fine grid finite element model confirmed the high strain concentration region and led to the incorporation of a steel plate as shown in figure 12 to provide the additional margin on strength. Based on the analysis and margin predicted with this design change, the FAA proceeded to address certification without requiring further ground tests and certification was granted in August 1982.

<u>L-1011 Composite Vertical Fin</u>.- The ground test plan for the L-1011 fin included limit load tests in three flight load conditions (i.e., shear, torsion, and bending) followed by a test to 106 percent of design ultimate load in bending, two lifetimes of fatigue tests to study damage growth, and finally residual strength test in bending. The 6 percent increase in ultimate load was imposed to account for the absence of moisture and temperature in the ground test series.

The L-1011 fin failed at 98 percent of design ultimate load during the planned test to 106 percent of design ultimate in bending. Failure caused separation of the cover and front spar along the entire length of the spar (see figure 13) as well as considerable internal damage to rib structure. After an investigation, the cause of failure was determined to be due to secondary loads, of which the principal contributor probably was local buckling of the cover near the front spar interface. While local buckling beyond limit load was allowed in the design, the influence of loads caused by buckling on the integrity of the structure was unexpected. Interlaminar tension forces caused delamination of the spar cap as shown by the insert in figure 13 and ultimate separation along the line of fasteners.

A post-failure study was conducted to assess the strength of the cover/spar design when subjected to stresses imposed by secondary loads causing interlaminar tension and transverse tension. Segments of the cover/spar design were tested as shown by figure 14. Secondary loads causing transverse tension could be induced by Poisson's effect and those causing interlaminar tension may be generated by local buckling of the cover or by rotation of the spar web caused by the method of load introduction. The first tests were on virgin material that had no prior loading and measured loads at failure indicated adequate margin. The estimated maximum interlaminar tension load expected in flight was 68 pounds per fastener. The second series of tests on undamaged segments of the failed spar, which had undergone several cycles of load, 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 an apparent result of a design weakness in the spar cap. In retrospect the 10 plies of zero degree oriented fibers (see figure 13) do not contribute to interlaminar strength and, in fact, provide the actual delamination plane. Interlaminar strength may have been enhanced if 45 degree plies or even 90 degree plies had been interspersed in the zero degree ply stack.

After the cause of failure was properly identified the ground test program was continued with a second fin. The composite fin was not to be certificated by FAA nor placed in flight service and redesign of the front spar and subsequent fabrication would have imposed unnecessary delays and cost to the program. Consequently, the second fin was reinforced to suppress the mode of failure of the first ground test article. The reinforcements, shown in figure 15, include aluminum doublers on the external surface of the cover, along the front and rear spar flanges, and at the rib attachment flange on the spar web. The primary purpose of the cover plate doubler was to provide extra material thickness so that runout flanges of the hat-stiffened cover (see figure 4) could be mechanically fastened in the area where the cover is only 10 plies thick.

The second fin was subjected to a series of damage tolerance tests which included lightning damage and field repair prior to final test for residual strength. Details of the damage tolerance program are shown in figure 16. The model was impacted at five locations on the surface to cause visible damage and then tested for one lifetime of spectrum fatigue loading. This was followed by simulated lightning damage consisting of vaporizing resin in the first four plies and a punch through as shown in figure 16. The component was then loaded in bending to limit load, repaired with an external patch as shown in the figure, subjected to a second lifetime of fatigue loads and then tested to failure.

The failed component, shown in figure 17, carried 119.7 percent of design ultimate load which was only slightly less than the predicted failure load of 121 percent. The mode of failure of the first unit was totally suppressed in this test and the failure did not propagate to other parts of the component. However, the failure mode included buckling of the cover as well as disbonding of some of the hat stiffeners. The stiffener disbond shown by NDI markings in figure 18 included flanges of two hat stiffeners and the disbond extended from the point of initiation at the front spar over most of the stiffener length to the root end. Again, secondary interlaminar tension loads at the stiffener/cover interface were of sufficient magnitude to cause failure. The influence of these loads may be a major design driver in future application of composites to primary structures.

DC-10 Composite Vertical Stabilizer.- Full scale tests of the DC-10 composite vertical stabilizer were selected to demonstrate adequate structural performance at critical flight conditions, and to verify compliance with FAA requirements for commercial flight certification. The test plan included vibration tests, static tests to design limit load, two fatigue lifetime spectra, and a fail-safe limit load test. For the static test, which began in December 1981, the full-size stabilizer was loaded in three flight critical design limit load conditions. Two limit load tests, inducing critical shear and critical torsion loading in the structure were successfully completed. The third limit load test, inducing critical bending in the structure, was applied to design limit load. While recording data at this load level, the structure failed. External damage included fractures of the compression cover and failure of the rear spar web. After removal of the cover, a detailed inspection revealed the failed spar and rib areas shown shaded in figure 19.

The failure investigation initially considered possible discrepancies in the following areas: design and analysis, test loads, material quality, and manufacturing. A preliminary analysis showed margins of safety in the structure to be adequate at design limit load. Loads data showed loading throughout the structure to agree with the planned test load distribution. Measured strains and deflections were found to be within design limits for the structure. Quality control records and inspection of failed parts revealed no correlation between part quality and observed failure modes. The investigation next focused on close examination of the failed areas identified in figure 19. The only failure found in the rear spar was a diagonal shear failure in the rear spar web in the bay immediately above the lower rudder actuator and a splice in the rear spar. A photograph of the spar web at this bay (figure 20) shows the failure extending diagonally across the web and through an access cutout.

The stabilizer was analyzed using a global finite element model to establish a failure scenario which would be consistent with all the failed elements on the structure. The failure sequence was established by deleting each structural member in the finite element model found to have a negative margin of safety as a result of a previous failure. When the rear spar was selected as the first failure the analysis indicated a progression of failures as shown by the numbered elements in figure 19. Each distinct failure observed in the structure was predicted in this manner and no other failure scenario produced the actual areas of failure.

The rear spar web is a thin sine-wave laminate, and contains access cutouts required to complete assembly of the structure and subsequent inspection and maintenance. There are seven such cutouts on the rear spar. Locations and details of the cutouts are shown in figure 21. These cutouts, about 4 by 5 inches in dimensions, are located in flat, reinforced areas in the spar web. Each cutout is fitted with a flat laminate cover which is attached to the web by bolts as shown in the diagram. This cover is designed to carry loads in the spar web as an integral part of the structure. Post-test examination of the spar web failure revealed a discrepancy in the bolt hole size both in the cover and in the spar web. These holes were 0.057 to 0.072 inches larger in diameter than the bolt.

To adequately analyze the sine-wave web rear spar area and the loose fit cover, a detailed finite element model was developed (figure 22). This model included fine grid detail of the spar cap, sine-wave web, flat web area, and cutout. The web cutout cover, not shown, was also modeled in detail. Loads were applied to this model while the web-to-cover interface was analytically varied to simulate various bolt fits. A reference case with no door installed was also run. Results of the analysis are shown in figure 23 where web shear failure loads and cutout perimeter strains are shown for three cases: (1) cover attached with tight holes; (2) cover attached with loose holes; and (3) cover off. The test failure shear load in the web, which was calculated from measured strain gage data, is also shown for comparison with calculated values. The spar web actually failed at a shear load somewhat higher than the calculated value for loose holes. The calculated load and strain values show that the cover with loose holes and the cutout without cover were almost equally ineffective in carrying required design load in the web. The cover attached with tight holes was predicted to provide adequate margin for failure load and to reduce strains at the hole perimeter to an acceptable level.

A test program was developed to verify the effect of cover fastener fit on the initial rear spar failure, and to evaluate the structural stability of the sine-wave web. These tests were also used to validate the detailed finite element model of the rear spar. A rib component having a sine-wave web and access cutout similar to the rear spar was used for the tests. The rib was mounted in the picture frame fixture shown in figure 24 and tested first without the cover, and then with the cover installed with oversize holes to match the rear spar configuration, and finally with a redesigned cover installed. The redesign cover, shown in figure 25, is flanged to provide added out-of-plane stiffness, and is permanently installed over the spar web cutout using an adhesive bond together with mechanical fasteners. The small hole in the redesigned cover will provide necessary access for inspection.

Results of the rib web shear test are shown in figure 26 where strains measured at the edge of the cutout (as shown in the inset diagram) are plotted as a function of web shear load. With the cover not installed, strains were extremely high and the test was stopped at a low load to avoid failing the specimen. With both the cover and web fastener holes drilled oversize to represent the ground test configuration, strain was somewhat reduced but still quite high. These results clearly show the importance of the fastener fit in transferring load across the shear web and reducing strains.

With the redesigned cover attached over the rib web cutout and adhesively bonded as well as mechanical fastened, the rib element was tested to failure. The strains for this test were much lower than for the other configurations and the web failed in the thin sine-wave section, well away from the reinforced cutout area. A detailed finite element analysis of the modified cover configuration agrees closely with the test data, confirming the adequacy of the finite element grid to represent the complex sine-wave web configuration and cutout cover. The results of the rib tests together with the analysis of the stabilizer and the rear spar web confirm the failure theory that the oversize holes in the cover reduced the ability of the cover to carry loads across the rear spar web, resulting in high stress concentration at the edge of the cutout.

Design modifications which have been incorporated on subsequent stabilizer units include, in addition to the redesigned covers for all cutouts, secondarily bonded ply reinforcements of selected internal spar and rib webs to increase their margins of safety (see reference 17). A second ground test article has now successfully completed all limit load tests and is scheduled to have completed fail-safe testing by July 1983.

Implications of Full Scale Ground Tests

One of the important features of the ACEE Composites Program is that it is recognized as a means to develop a level of understanding and confidence in the performance of composites for application to transport structures, and, consequently, structural failures of full scale components are unique opportunities to identify failure modes peculiar to composite structures. The medium-primary program provided opportunities for the composites industry to develop a keener insight into requirements for design, manufacturing, and testing of composites which should not only enhance understanding but should identify requirements for developing a data base which will assure the "safety of flight" already established for metals. The following discussion will review some of these insights, which although are not necessarily new, were manifested with this ground test experience.

The designers of composite structures lack the extensive "standard practice" foundation that accompanies metal designs and, consequently, they must incorporate considerable intuitive knowledge until adequate design guides are developed and validated. 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 can be safely ignored in metal structures, the sensitivity of current composite materials to interlaminar forces can lead to serious weaknesses being overlooked in the design of composite structures. Such loads may be produced by eccentricities, irregular shapes, stiffness changes, and discontinuities, and their effects are magnified by the brittle nature of composites, which precludes load redistribution associated with plasticity effects. Unfortunately, detailed problems in composites require fine-grid finite element models which are frequently complex to generate and expensive to run. However, for strength critical structures selective finite element modeling is essential to identify potential problems early in the design phase. The real challenge may be in selecting the areas to analyze. Areas for consideration should include regions of intense load gradients and regions of unusual structural complexity.

Criteria for assembly of composite structures are generally more demanding than those in metals. The nonyielding aspects of composites makes the determination of load distributions in mechanically fastened joints and the redistribution of loads difficult. Since mechanical fasteners are still the primary method of final assembly, the quality of drilled holes and control of hole tolerance are critical in the assembly process and can be key factors in performance.

Simulation of aerodynamic and inertia loads on full scale composite test articles is always a problem due to the necessity to apply loads at discrete points. Careful consideration must be given to whether the representation of distributed loads permits proper and adequate interrogation of the composite structure. Finesse is required for load introduction to insure that unacceptable secondary loads are not induced. On the other hand, load introduction schemes may mask real secondary failure modes.

Subcomponent tests should not be used as the exclusive method of full scale validation because of secondary and off-angle loads introduced into a built-up structure during deflection, while under load, which may not exist in a subcomponent test. The basic problem is the difficulty in duplicating important details, such as edge restraint and loading, that are required to maintain a functional relationship between the subcomponent and full scale structure.

Two factors which appear critical to the widespread use of composites in aircraft structures are the essentially elastic stress-strain nature of composites to ultimate failure load and their susceptibility to failure in interlaminar tension and shear. Work is in progress to evolve a composite material system with improved ductility and interlaminar toughness, and yet retains desirable features such as adequate mechanical properties, processability, environmental stability, and solvent resistance (see reference 18).

Concluding Remarks

The major transport manufacturers have undertaken the development of technology required for the application of composites to empennage structure of large transport aircraft. Structural components which have been designed and manufactured as direct replacements for existing metallic parts have demonstrated weight reductions of more than 22 percent. In the ground test programs, the performance of full scale composite components was assessed for various static loads simulating critical flight conditions, damage tolerance conditions, spectrum fatigue loadings, and fail-safe conditions. Each of three ground test articles failed at loads less than expected, but detailed investigations of the failures identified the cause of failure and failure sequence.

The ground test results provided insight into a number of problems that must be addressed before composites can be successfully applied to primary structure. The brittle nature of composites and their relative weakness in interlaminar tension and shear will be major design concerns until at least partially alleviated by material improvements. These features were instrumental in each of the early failures and in two of the three components the failure modes were not evident from subcomponent tests. Design modifications have been made on all three structural components and two of the components have successfully completed all ground tests.

References

1. Vosteen, Louis F., "Composite Structures for Commercial Transport Aircraft," NASA TM 78730, June 1978.

2. Leonard, Robert W., "Fuel Efficiency Through New Airframe Technology," NASA TM 84548, August 1982.

3. Dexter, H. Benson, "Composite Components on Commercial Aircraft," NASA TM 80231, March 1980.

4. Chovil, D. V., Desper, O. E., Harvey, S. T., Jamison, E. S., McCarty, J. E., and Syder, H., "Advanced Composite Elevator for Boeing 727 Aircraft Volume 1 - Technical Summary," NASA CR 3290, January 1981.

5. Cominsky, A., et al., "Manufacturing Development of DC-10 Advanced Rudder," NASA CR 159060, August 1979.

6. Griffin, C. F. and Dunning, E. G., "Development of an Advanced Composite Aileron for the L-1011 Transport Aircraft," NASA CR 3517, February 1982.

7. Peterson, D. C., Pritchett, L. D., Wogulis, B. R., Wilson, D. R., Parsons, J. T., Aniversario, R. B., Harvey, S. T., and McCarty, J. E., "Technical Summary - Design, Ancillary Testing, Analysis, and Fabrication Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft," NASA CR 3648, February 1983.

8. Peterson, D. C., Pritchett, L. D., Wogulis, B. R., Wilson, D. R., Parsons, J. T., Aniversario, R. B., Harvey, S. T., and McCarty, J. E., "Technical Summary - Non-ancillary Testing, Production, and Cost Analysis Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft," NASA CR 3649, February 1983.

9. Peterson, D. C., Pritchett, L. D., Wogulis, B. R., Wilson, D. R., Parsons, J. T., Aniversario, R. B., Harvey, S. T., and McCarty, J. E., "Final Report - Design, Ancillary Testing, Analysis, and Fabrication Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft," NASA CR 166011, February 1983.

10. Peterson, D. C., Pritchett, L. D., Wogulis, B. R., Wilson, D. R., Parsons, J. T., Aniversario, R. B., Harvey, S. T., and McCarty, J. E., "Final Report - Non-ancillary Testing, Production, and Cost Analysis Data for the Advanced Composite Stabilizer for Boeing 737 Aircraft," NASA CR 166012, February 1983.

11. McCarty, J. E. and Wilson, D. R., "Advanced Composite Stabilizer for Boeing 737 Aircraft," paper presented at Sixth DOD/NASA Conference on Fibrous Composites in Structural Design," New Orleans, Louisiana, January 24-27, 1983.

12. Ary, A., Axtell, C., Fogg, L., Jackson, A., James, A., Mosesian, B., Vanderwier, J., and Van Hamersveld, J., "Flight Service Evaluation of an Advanced Composite Empennage Component on Commercial Transport Aircraft," Phase I - Final Report: Engineering Development, NASA CR 144986, May 1976.

13. Jackson, A. C., Crocker, J. F., Ekvall, J. C., Eudaily, R. R., Mosesian, B., Van Cleave, R. R., and Van Hamersveld, J., "Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft," Phase II - Final Report Design and Analysis, NASA CR 165634, April 1981.

14. Alva, F., Brozovic, G., Carll, B., Eudaily, R., Henkel, J., Jackson, A., Johnston, R., Mosesian, B., and O'Brien, R., "Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft," Phase IV - Final Report: Manufacturing Development, NASA CR 165885, May 1982.

15. Jackson, A., and Dorward, F., "Advanced Manufacturing Development of a Composite Empennage Component for L-1011 Aircraft," Phase V - Final Report: Full Scale Ground test, NASA CR 166015, December 1982.

16. Jackson, A. C., "Testing of the L-1011 Advanced Composite Vertical Fin," paper presented at Sixth Conference on Fibrous Composites and Structural Design, New Orleans, Louisiana, January 24-27, 1983.

17. Palmer, J. M., Jr., and Stephens, C. O., "DC-10 Composite Vertical Stabilizer Ground Test Program," Sixth Conference on Fibrous Composites in Structural Design, New Orleans, Louisiana, January 24-27, 1983.

18. Bohon, H. L., "Advanced Materials Technology," Opportunities for Composites in Commercial Transport Structures, NASA CP 2251, November 16-17, 1982.

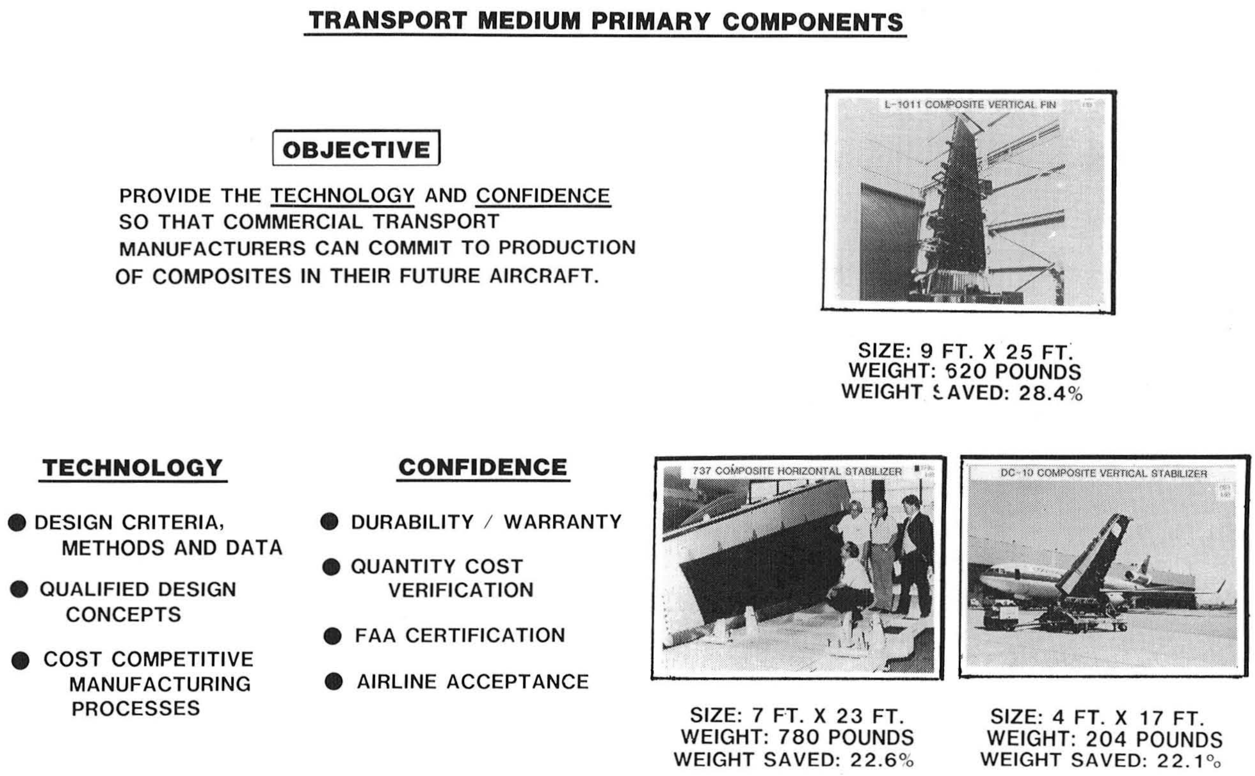


Figure 1 - Composite primary aircraft structures in the NASA aircraft energy efficiency (ACEE) program.

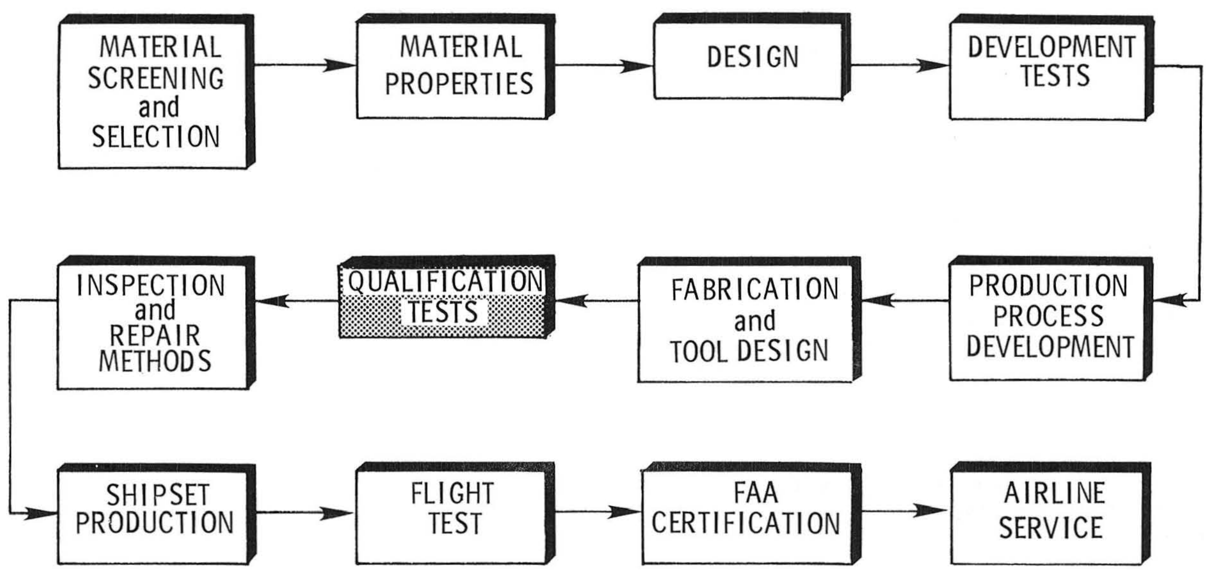


Figure 2 - Medium primary component development program.

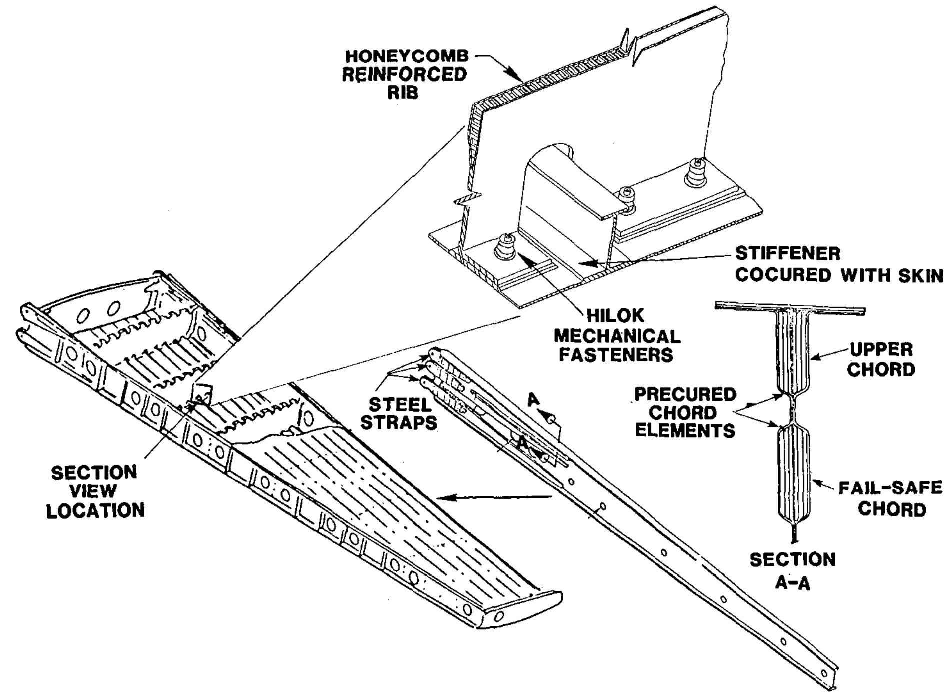


Figure 3 - 737 composite horizontal stabilizer - assembly methods.

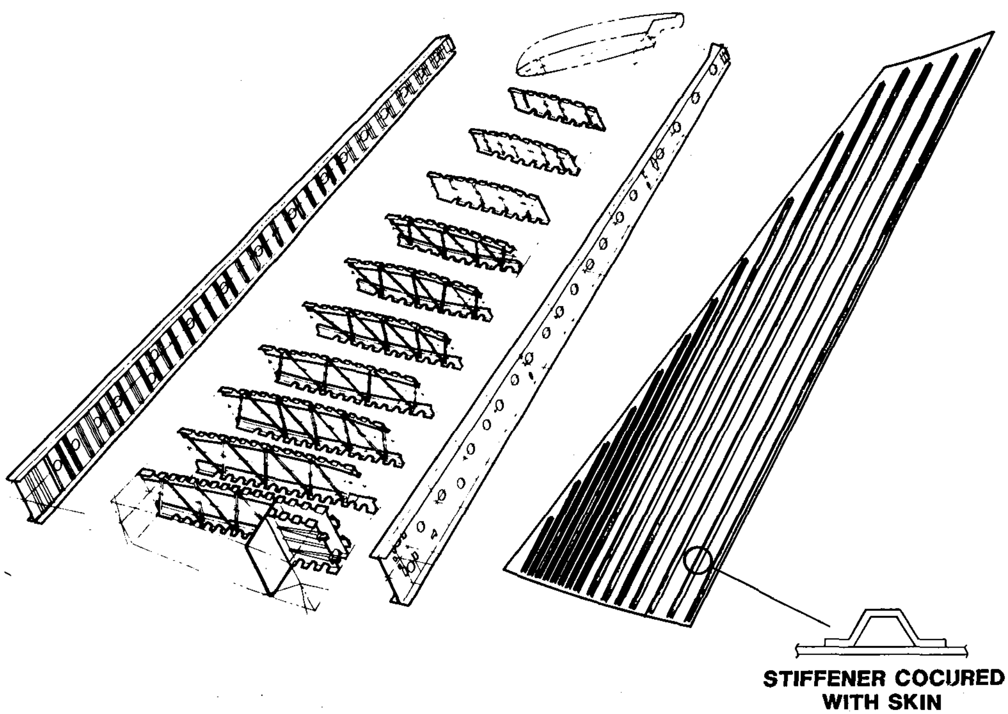


Figure 4 - L-1011 composite vertical fin - structural configuration.

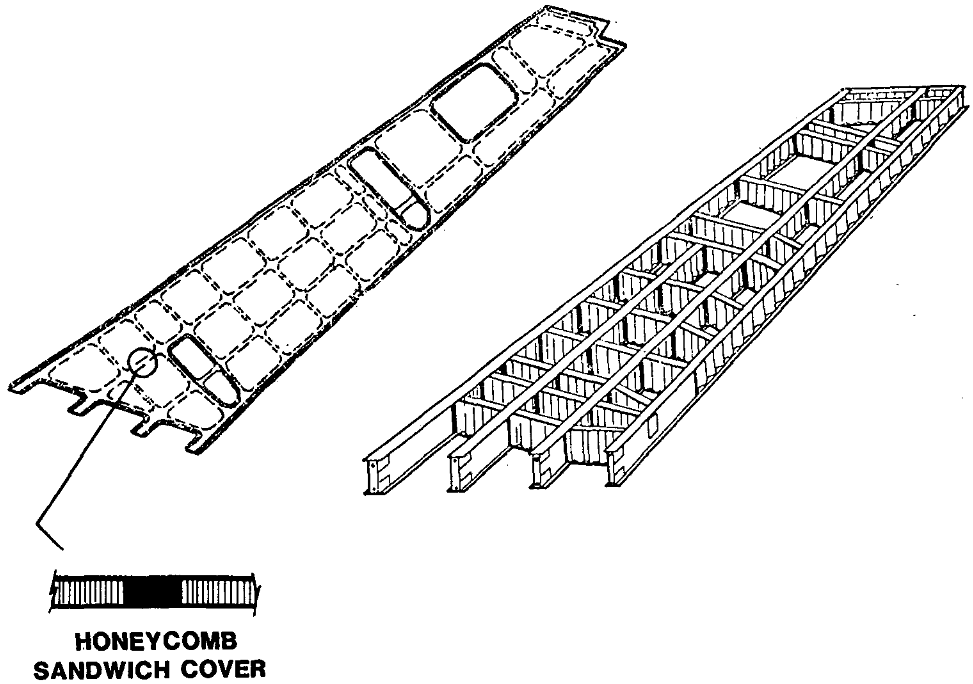


Figure 5 - DC-10 composite vertical stabilizer - structural configuration.

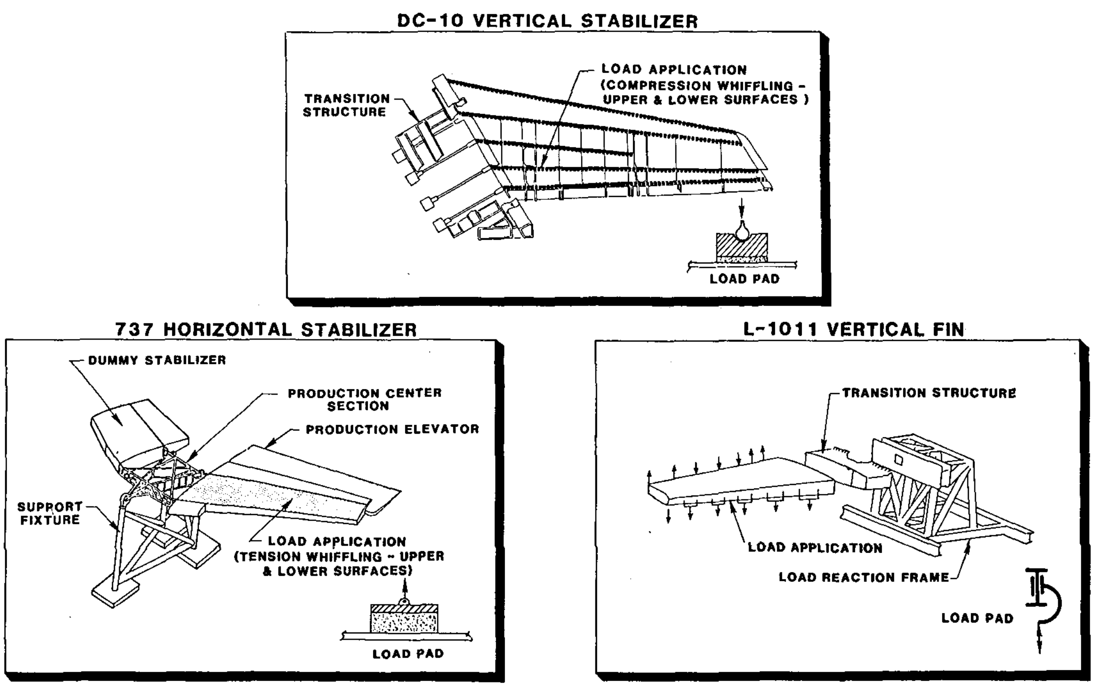


Figure 6 - Full-scale ground test load techniques.

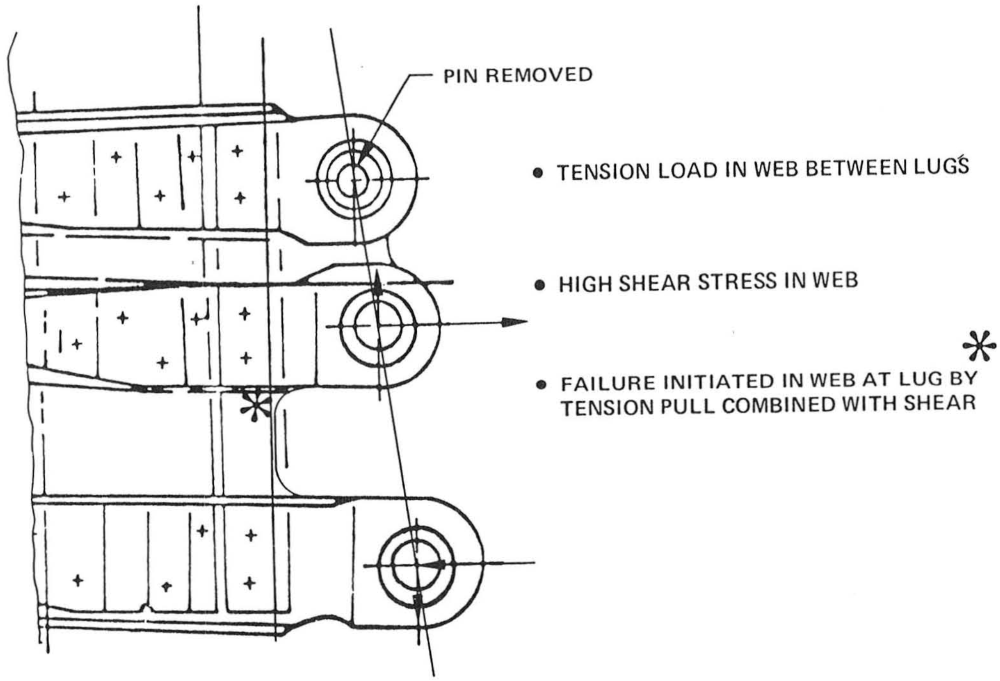


Figure 7 - 737 composite horizontal stabilizer - fail-safe test configuration.

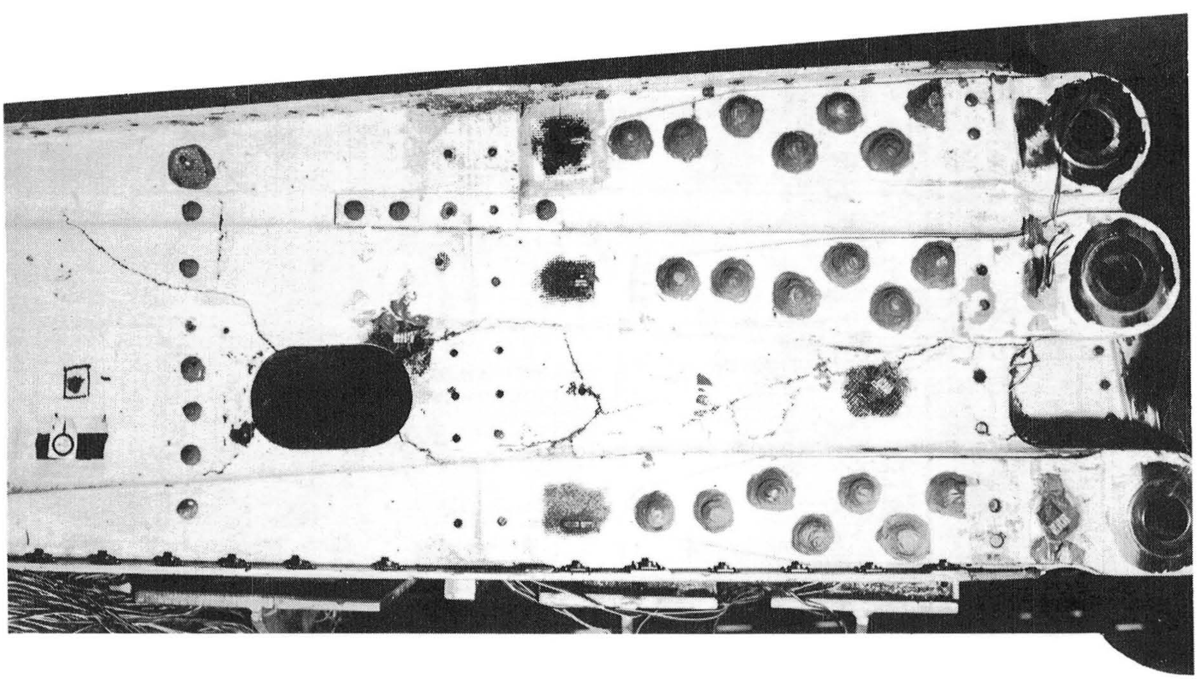


Figure 8 - 737 composite horizontal stabilizer - rear spar failure of ground test article.

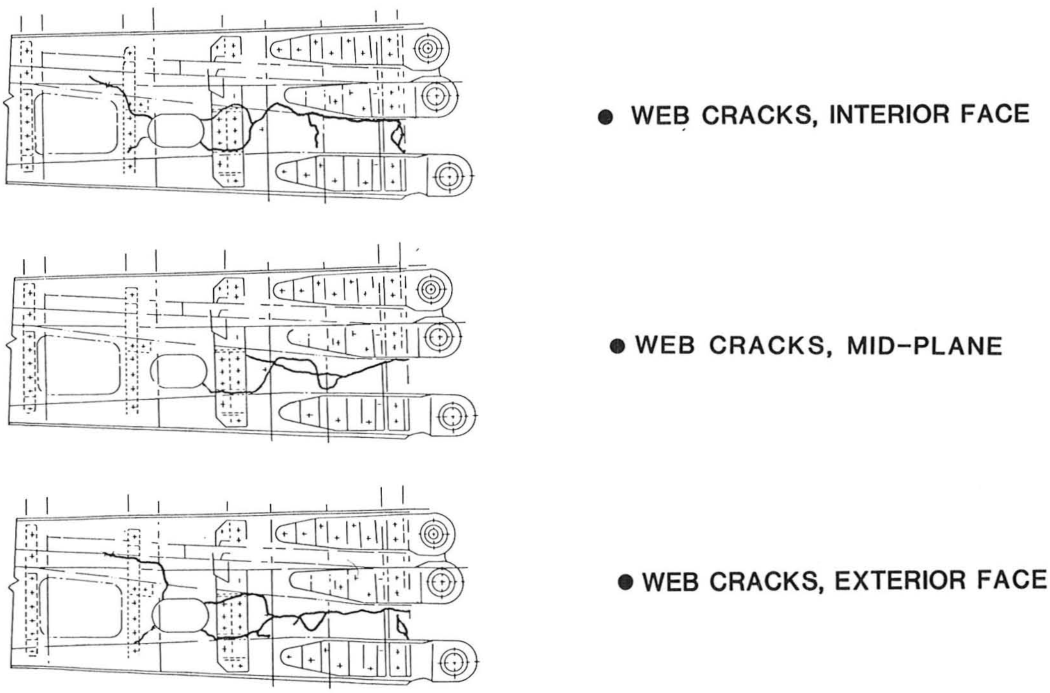


Figure 9 - 737 composite horizontal stabilizer - crack propagation in ground test article.

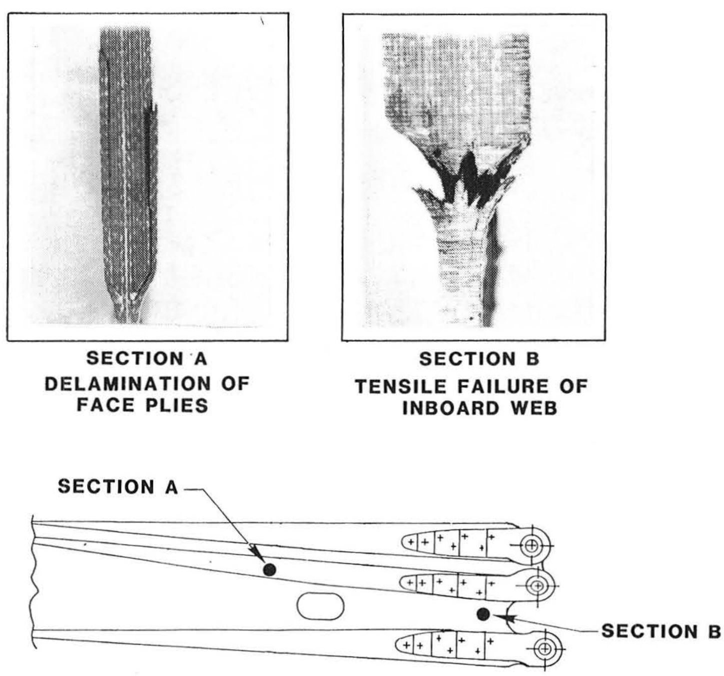


Figure 10 - 737 composite horizontal stabilizer - rear spar web failure.

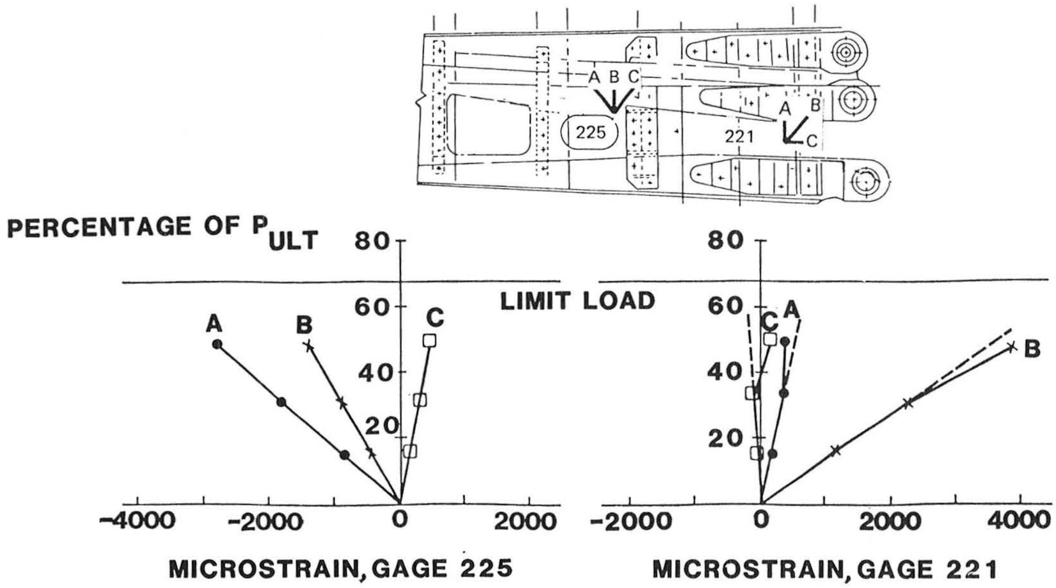


Figure 11 - 737 composite horizontal stabilizer - rear spar strain data.

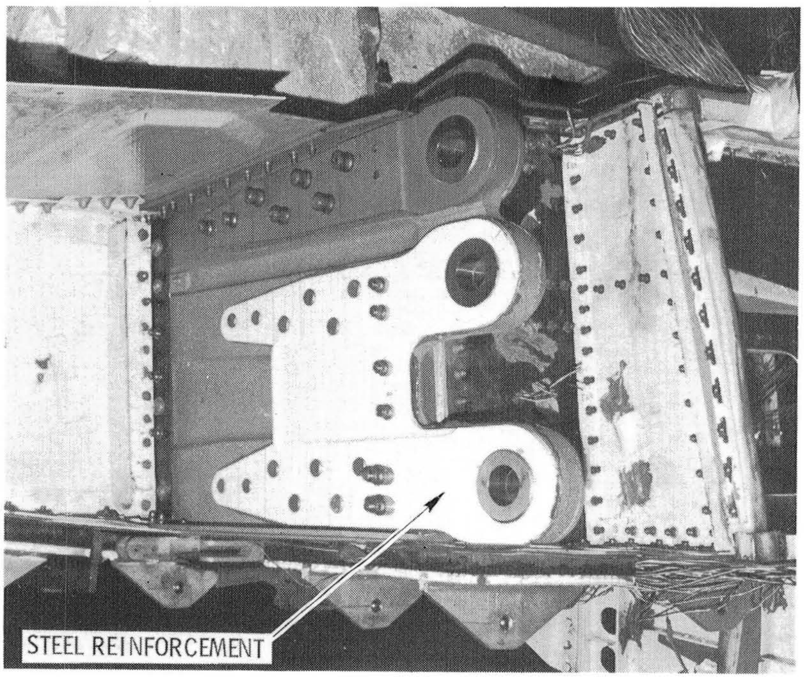


Figure 12 - 737 composite horizontal stabilizer - steel reinforcement of rear spar.

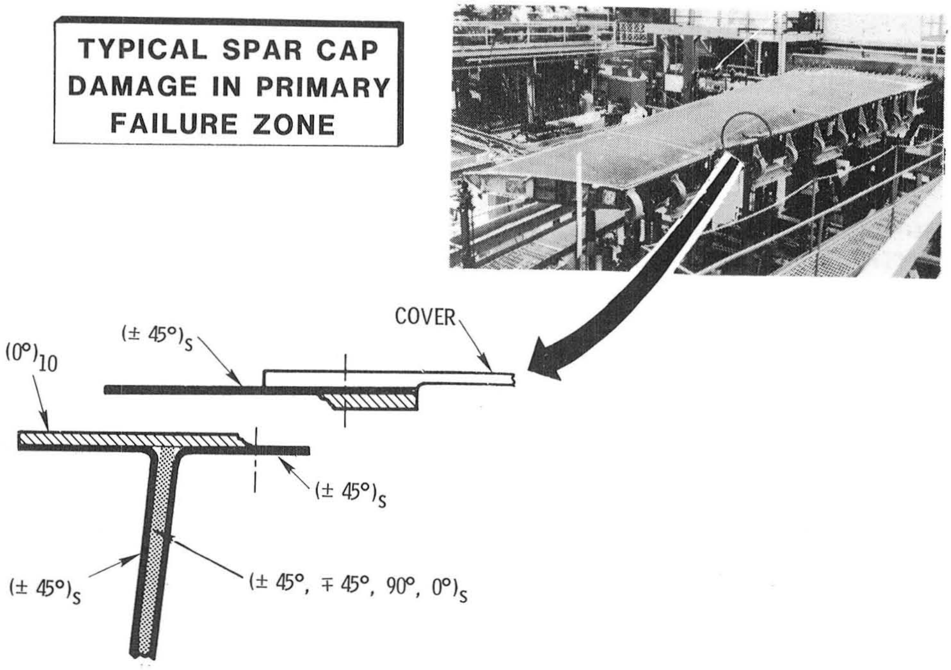
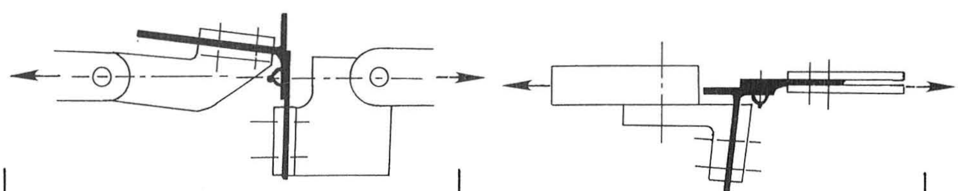


Figure 13 - L-1011 composite vertical fin static test failure at 98% design ultimate load.



SPECIMEN CONDITIONING	INTERLAMINAR 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

Figure 14 - L-1011 composite vertical fin - influence of load cycling on interlaminar strength.

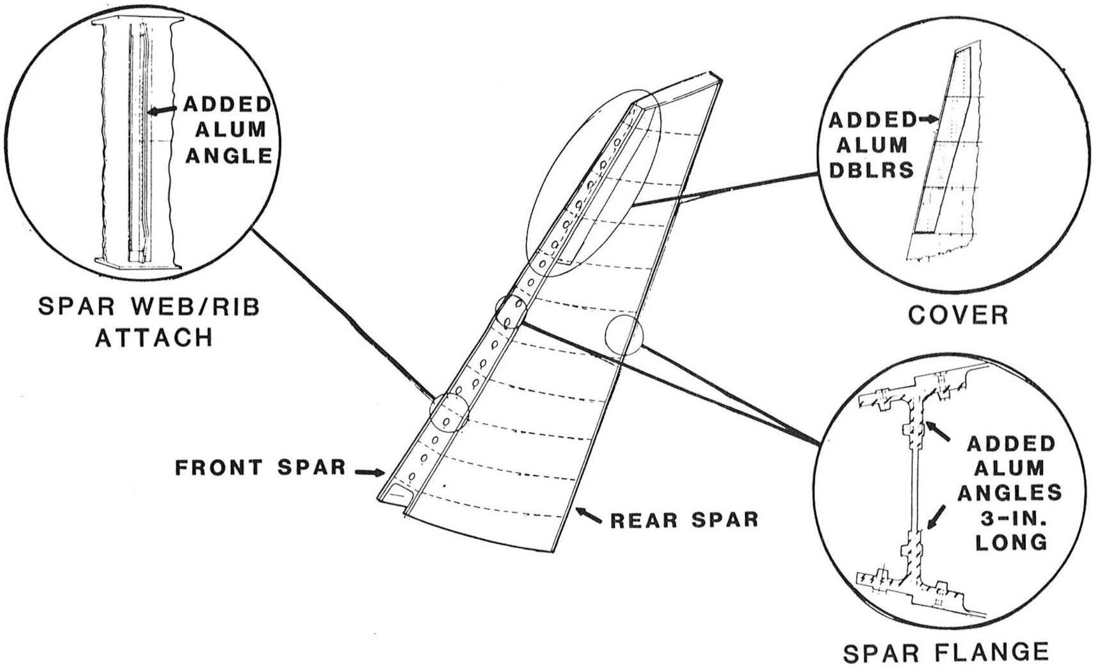


Figure 15 - L-1011 composite vertical fin -
ground test article modifications.

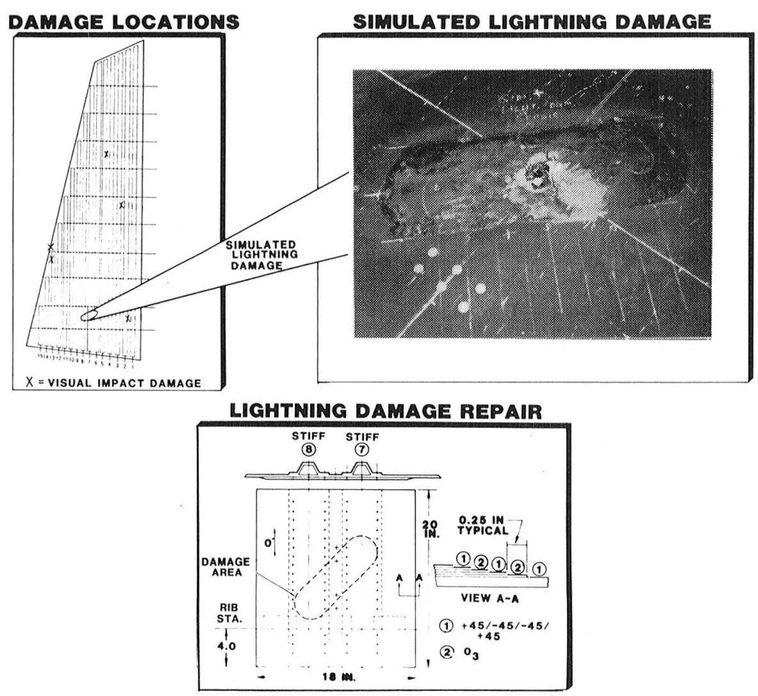


Figure 16 - L-1011 composite vertical fin -
damage and repair.

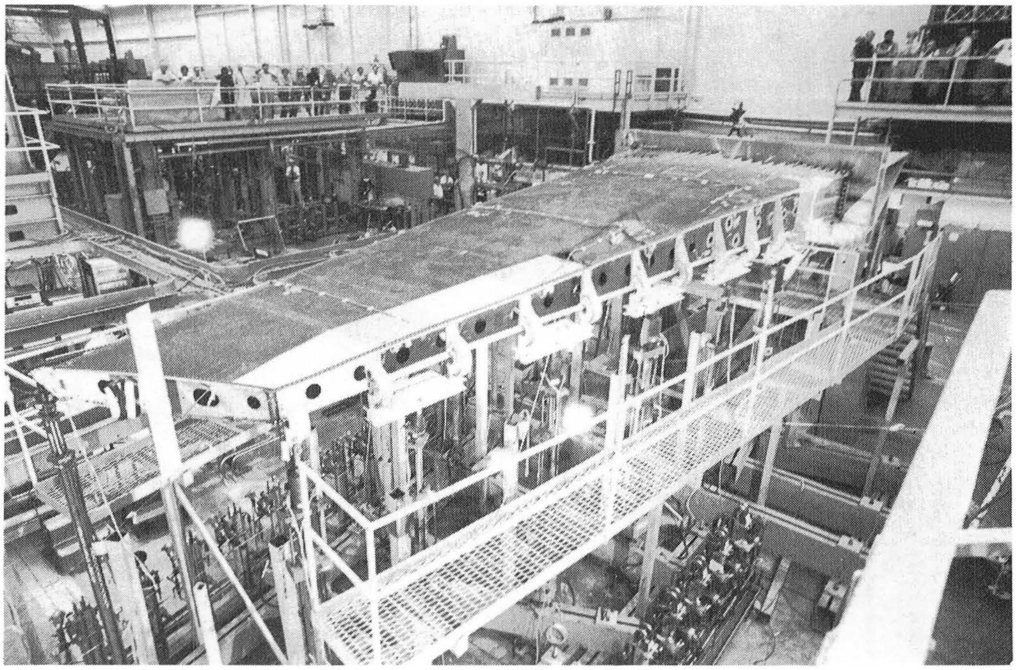


Figure 17 - L-1011 composite vertical fin -
static test failure at 119.7% design ultimate load.

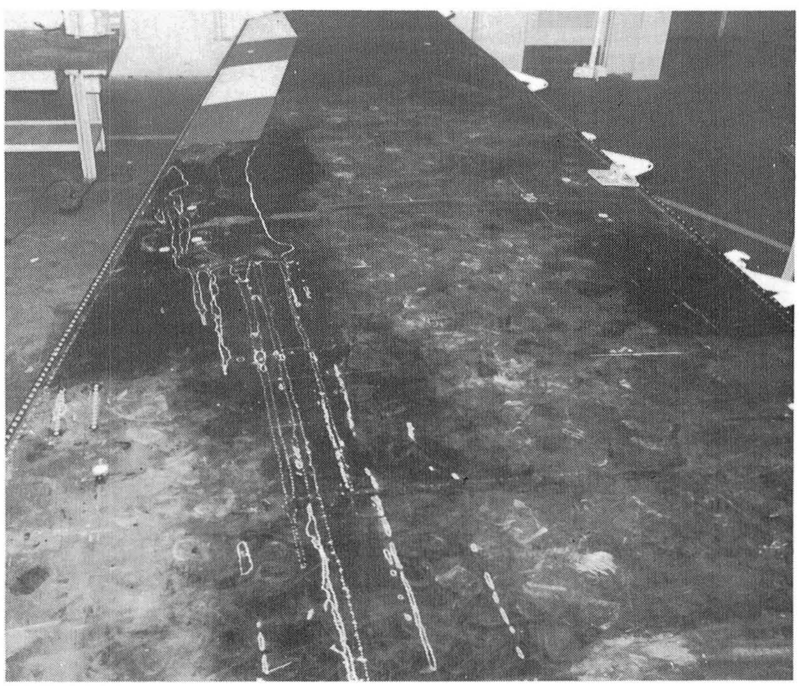


Figure 18 - L-1011 composite vertical fin -
post-failure NDI.

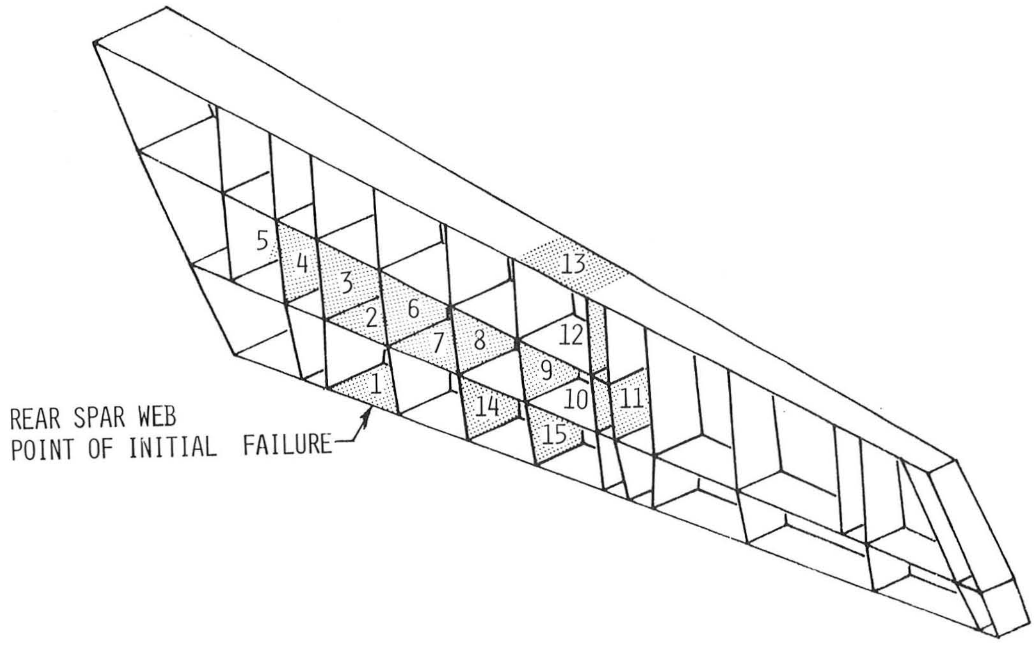


Figure 19 - DC-10 composite vertical stabilizer - ground test failure sequence.

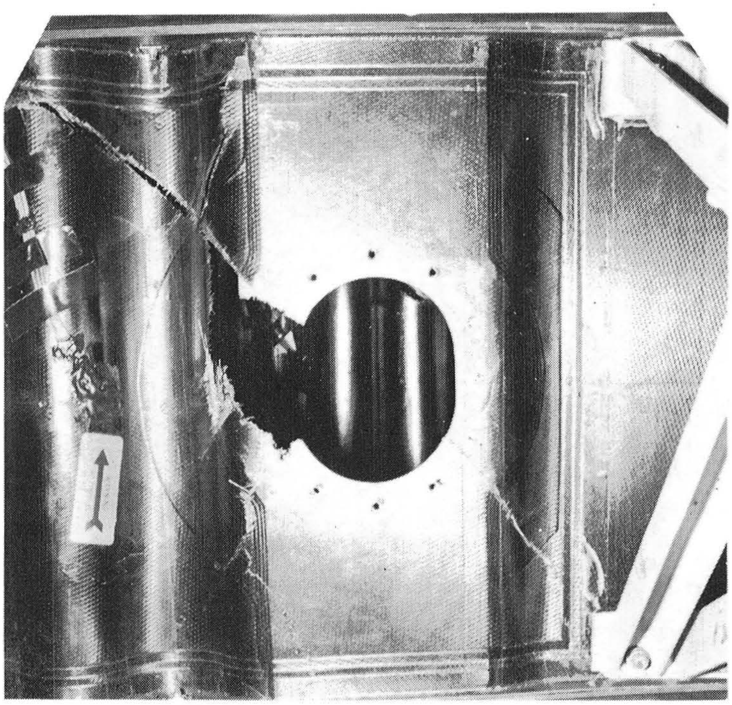


Figure 20 - DC-10 composite vertical stabilizer - rear spar static test failure at limit load.

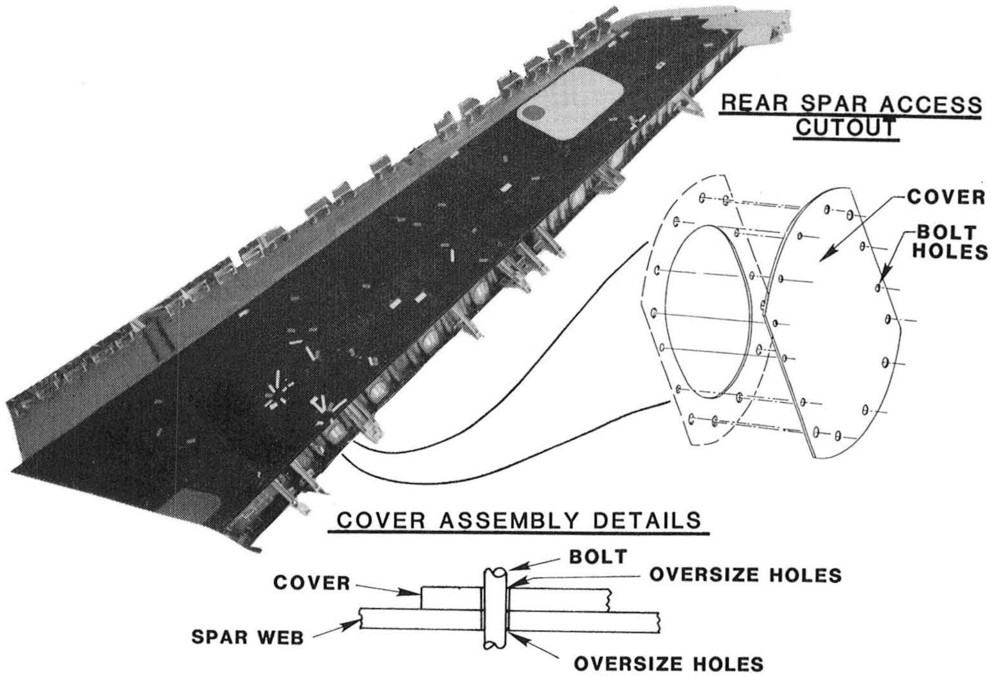


Figure 21 - DC-10 composite vertical stabilizer - failure initiation at rear spar access cutout.

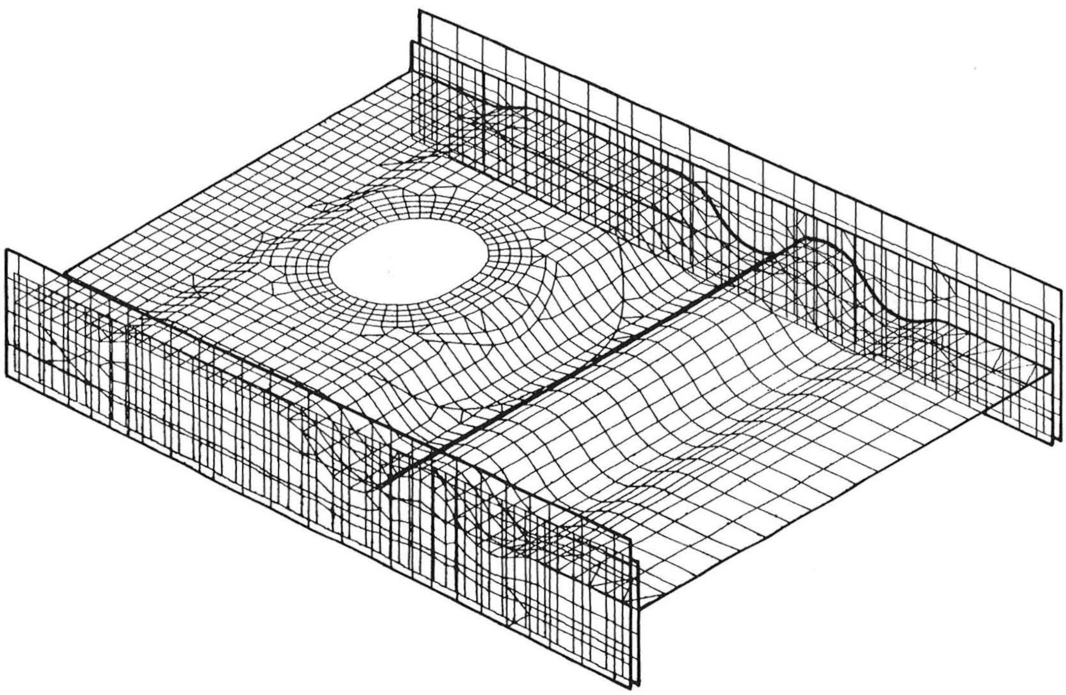


Figure 22 - DC-10 composite vertical stabilizer - rear spar finite element model.

TEST FAILURE LOAD	ANALYSIS FAILURE LOADS FINITE ELEMENT MODEL
832 LB/IN	COVER ON, TIGHT HOLES - 1140 LB/IN COVER ON,LOOSE HOLES - 620 LB/IN COVER OFF - 570 LB/IN

CALCULATED STRAINS AT EDGE OF CUTOUT, μIN/IN

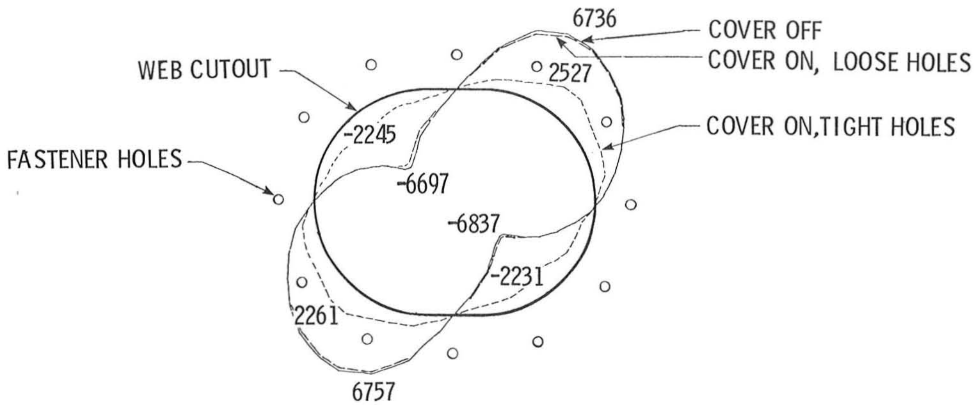


Figure 23 - DC-10 composite vertical stabilizer - rear spar web strains and shear loads.

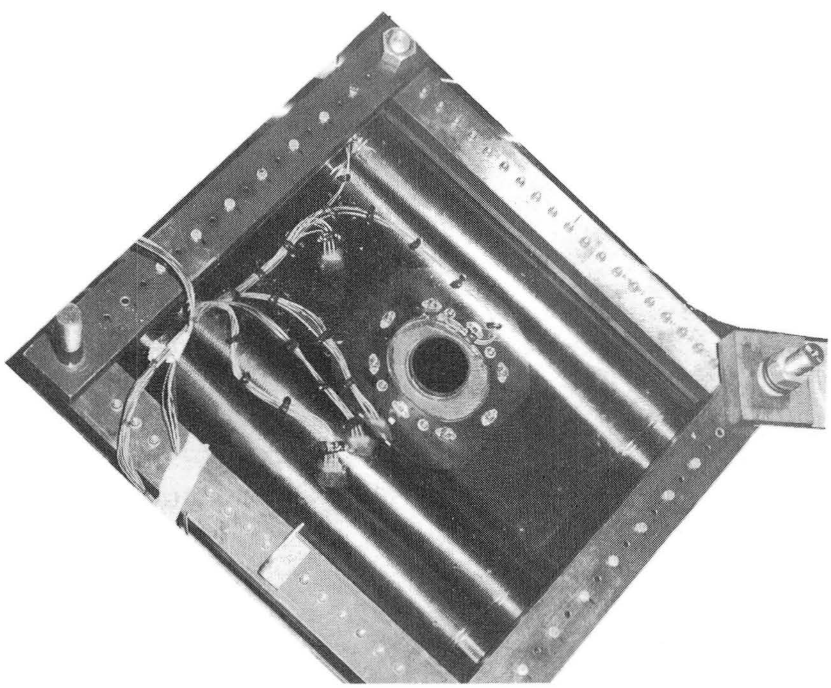


Figure 24 - DC-10 composite vertical stabilizer - shear web test article.

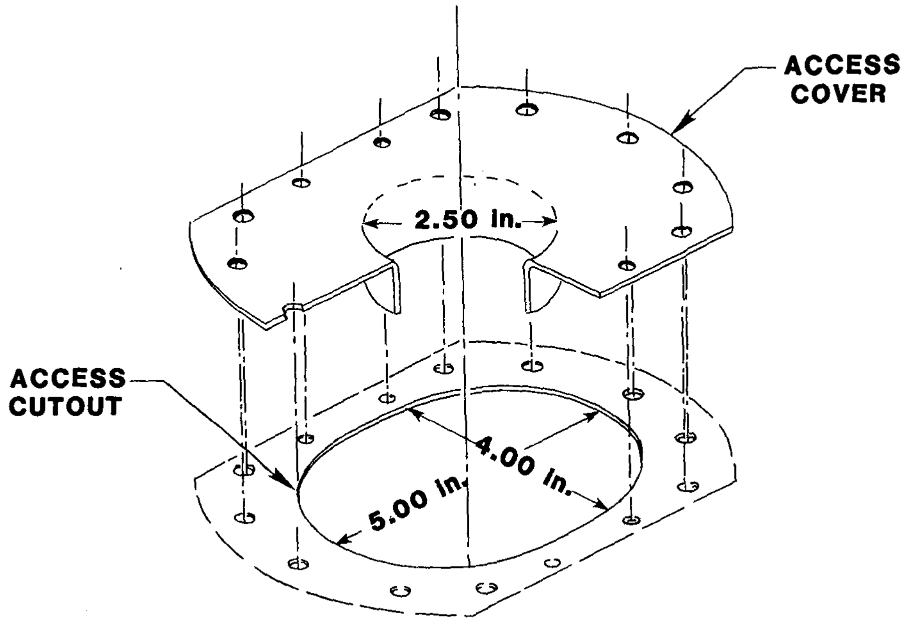


Figure 25 - DC-10 composite vertical stabilizer - shear web cutout cover redesign.

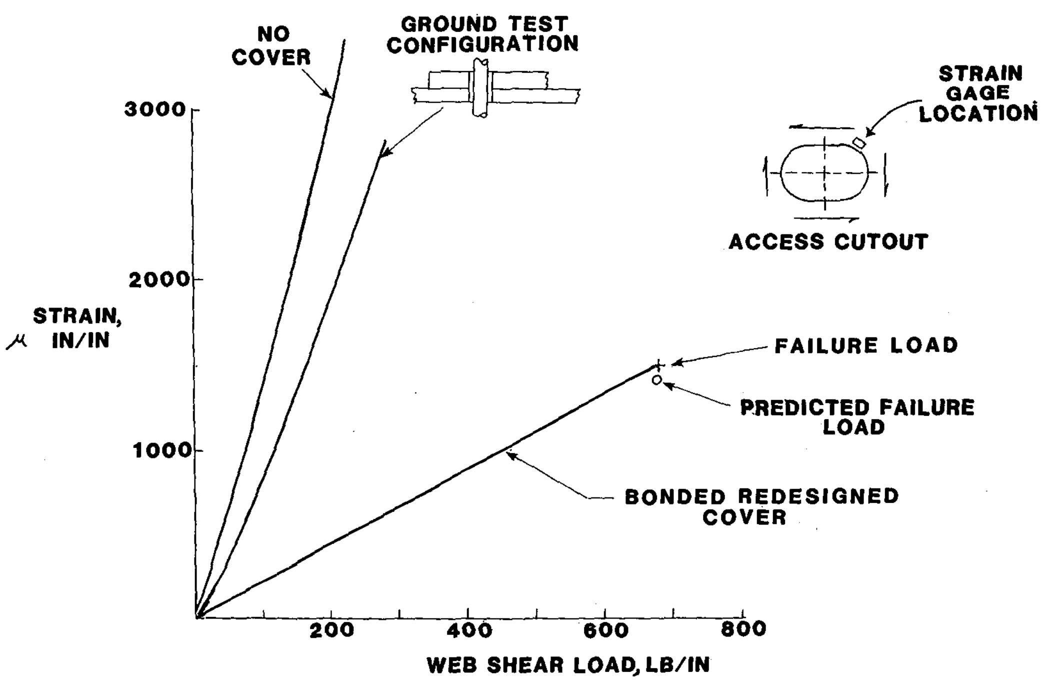


Figure 26 - DC-10 composite vertical stabilizer - access cutout strain data.

1. Report No. NASA TM 84627	2. Government Accession No.	3. Recipient's Catalog No.

4. Title and Subtitle Ground Test Experience With Large Composite Structures for Commercial Transports	5. Report Date March 1983
	6. Performing Organization Code 534-03-13
7. Author(s) H.L. Bohon, A.J. Chapman, III, H.A. Leybold	8. Performing Organization Report No.
	10. Work Unit No.
9. Performing Organization Name and Address NASA Langley Research Center Hampton, Virginia 23665	11. Contract or Grant No.
	13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, DC 20546	14. Sponsoring Agency Code

15. Supplementary Notes

For Presentation at: March 23-25, 1983
AMERICAN HELICOPTER SOCIETY COMPOSITE STRUCTURES SPECIALIST'S CONFERENCE - Philadelphia, Pennsylvania

16. Abstract

Composite technology applicable to transport empennage structures has been developed through contracts sponsored by the NASA Aircraft Energy Efficiency (ACEE) Program Office. The empennage components, the horizontal stabilizer for the Boeing 737, the vertical fin for the Lockheed L-1011, and the vertical stabilizer for the Douglas DC-10, have been designed to replace the existing metal structures without modification of other parts of the aircraft. A principal element in the program for each component is an extensive ground test series of a full size structure. The programs for two of the components, the 737 Horizontal Stabilizer and the DC-10 Vertical Stabilizer, include, as objectives, FAA certification and airline flight service. The ground test series for these components are, consequently, essential steps in verifying compliance with FAA certification requirements. The initial ground test of each component resulted in structural failure at less than ultimate design loads. While such failures represent major program delays, the investigation and analysis of each failure revealed significant lessons for effective utilization of composites in primary structure. Foremost among these are secondary loads that produce through-the-thickness forces which may lead to serious weaknesses in an otherwise sound structural design. The sources, magnitude, and effects of secondary loads need to be thoroughly understood and accounted for by the designers of composite primary aircraft structures.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Composites, Materials, Primary Aircraft Structure, Full Scale Testing, Transport Aircraft, Graphite Epoxy	Unclassified - Unlimited Subject Category 39

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
Unclassified	Unclassified	21	A02

For sale by the National Technical Information Service, Springfield, Virginia 22161