RESIDUAL-STRENGTH TESTS OF L-1011 VERTICAL FIN COMPONENTS AFTER 10 AND 20 YEARS OF SIMULATED FLIGHT SERVICE

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INTRODUCTION

A major issue facing the Aerospace Industry today is the effect of flight service exposure on the performance of large graphite-epoxy aircraft structural components. It is essential that these composite structures survive extensive cyclic loading and environmental conditioning of sufficient severity to develop the necessary confidence for flight service durability.

Part of the NASA/ACEE sponsored Production Readiness Verification Test (PRVT) plan (refs. 1, 2) for the Lockheed Advanced Composite Vertical Fin (ACVF) program was to determine the effect of long-term durability testing on the residual strength of graphite-epoxy cover panel and spar components of the Lockheed L-1011 aircraft vertical stabilizer.

Lockheed designed and fabricated 22 cover panels and 22 spar components. Ten of each of the components were statically tested to failure to determine their static strength. The results of these tests established the strength variability and production quality of the original components. The remaining 12 cover panels and 12 spar specimens were subjected to long-term durability testing which simulated 10 and 20 years of flight service exposure.

To determine the effect of simulated flight testing exposure on the residual strength of the cover panel and spar specimens, NASA Langley Research Center performed eleven static tests to failure on these specimens. This paper presents the results of these residual-strength tests. The structural behavior and failure mode of both cover panel and spar specimens are addressed, and the test results obtained are compared with the static test results obtained by Lockheed. In addition, the effect of damage on one of the spar specimens is described.

PRVT DURABILITY TESTING

The long-term durability testing of 12 cover and 12 spar specimens was conducted by the Lockheed-California Company (CALAC). These structural components were placed in environmental chambers and subjected to several test spectra. A spectrum of temperature and humidity and a cyclic load spectrum were applied. A brief description of the durability test program is shown below. Several spar and cover specimens were subjected to half a life cycle, or the equivalent of 10 years of simulated flight service. The other specimens were subjected to a full 20-year life cycle spectrum. (See fig. 1.)

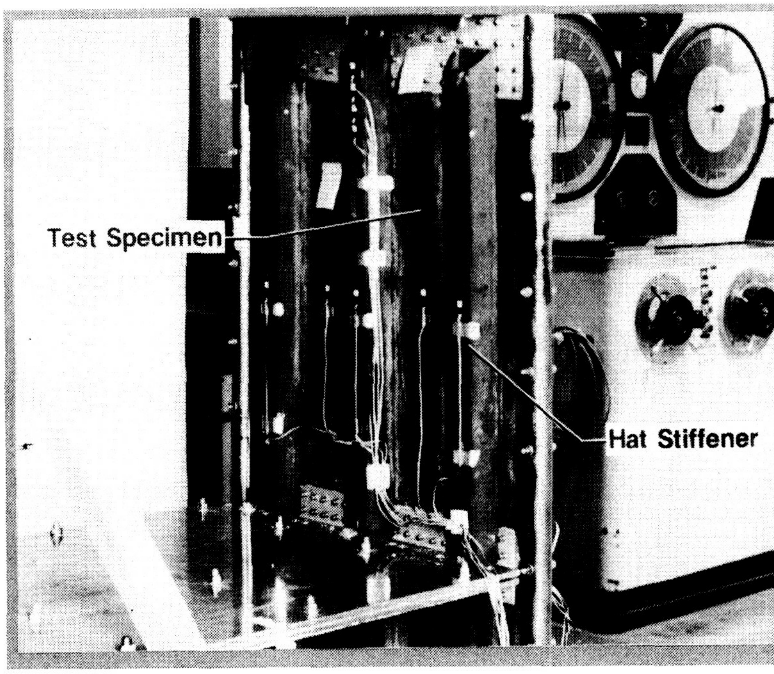
- Environment
  - Temperatures: -30ºF to 140ºF
  - Humidity: 0 to 100%
- Cyclic Loading
  - Spectrum Fatigue
  - 20 Year Life (36000 Flights)
  - 6.2 Flight/Thermal Cycle

Figure 1

STIFFENED COVER PANEL SPECIMENS

A typical stiffened cover panel (refs. 1, 3) constructed from T300/5208 graphite-epoxy tape is shown in figure 2. The cover panel specimen measures five feet in length and two feet in width. The upper half, or top bay, of the cover panel specimen is shown in this figure. The cover panel specimen consists of three hat stiffeners which are co-cured with the skin. The panel skin is composed of 0° and $\pm 45^{\circ}$ layers and tapers in thickness from 34 plies at the top bay to 16 plies at the bottom bay.

CALAC designed and constructed 22 of these cover panel specimens. The design of these cover panel specimens was based on a stiffness criterion which required the graphite-epoxy panel to match its metal counterpart. Twelve of these specimens were subjected to durability tests (conditioned) and ten were statically tested to failure. Six of the conditioned cover panel specimens were tested at NASA Langley to assess the effects of long-term durability testing on residual strength. Two 20 year conditioned specimens were loaded, during one cycle, to design ultimate load (54600 lb) and held at this load for a few seconds.



Specimen Conditions	Number Tested
Control (Calac Tests)	10
10 Years of Conditioning	2
20 Years of Conditioning	2
20 Years of Conditioning at High Strain	2

Figure 2

TEST SET-UP FOR STIFFENED COVER PANEL SPECIMENS

The test set-up for the stiffened cover panel specimens is shown in figure 3. The test set-up consists of two steel I-beams which form the reaction frame, supporting three aluminum alloy plates which simulate rib structures. The frame also provides stability while the cover panel specimen is loaded in compression. There are four steel link bars and two aluminum angle bars which connect the cover panel specimen to the frame and carry the reaction loads from the specimen. The cover panel is fixed at both ends by the two end fixtures and subjected to compression loading in the NASA Langley one million pound testing machine.

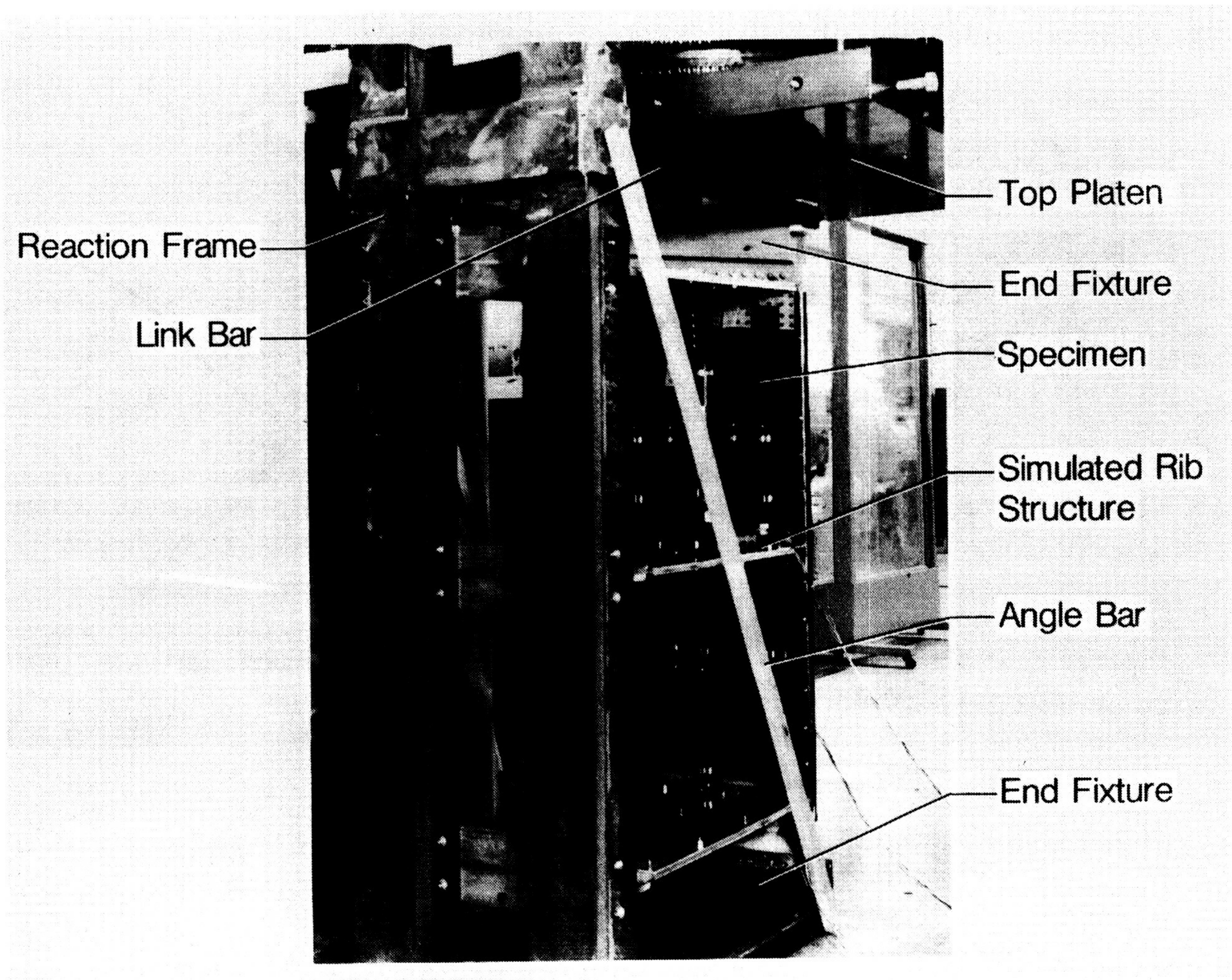


Figure 3

TYPICAL RESULTS FOR COVER PANEL SPECIMENS

Some typical results for the cover panels are shown in figure 4. On the left is a photograph of a typical moire fringe pattern showing the buckling mode of the lower 16-ply-thick bay of the cover panel specimen. Large amounts of deformation occur in the skin between stiffeners as indicated by the closely space fringes.

Plotted on the right are some typical back-to-back strain gage results. The plots for gages A and B (located on the skin between stiffeners) indicate strain gage reversal at 55000 lb, which is the onset of buckling. Buckling occurred above the design limit load of 30000 lb. The specimen is loaded well into the post-buckling range and fails at 92000 lb. Gages C and D, located respectively on the cap and skin sides of the stiffener, show that some bending occurs in the stiffener, but is not as pronounced as in the skin.

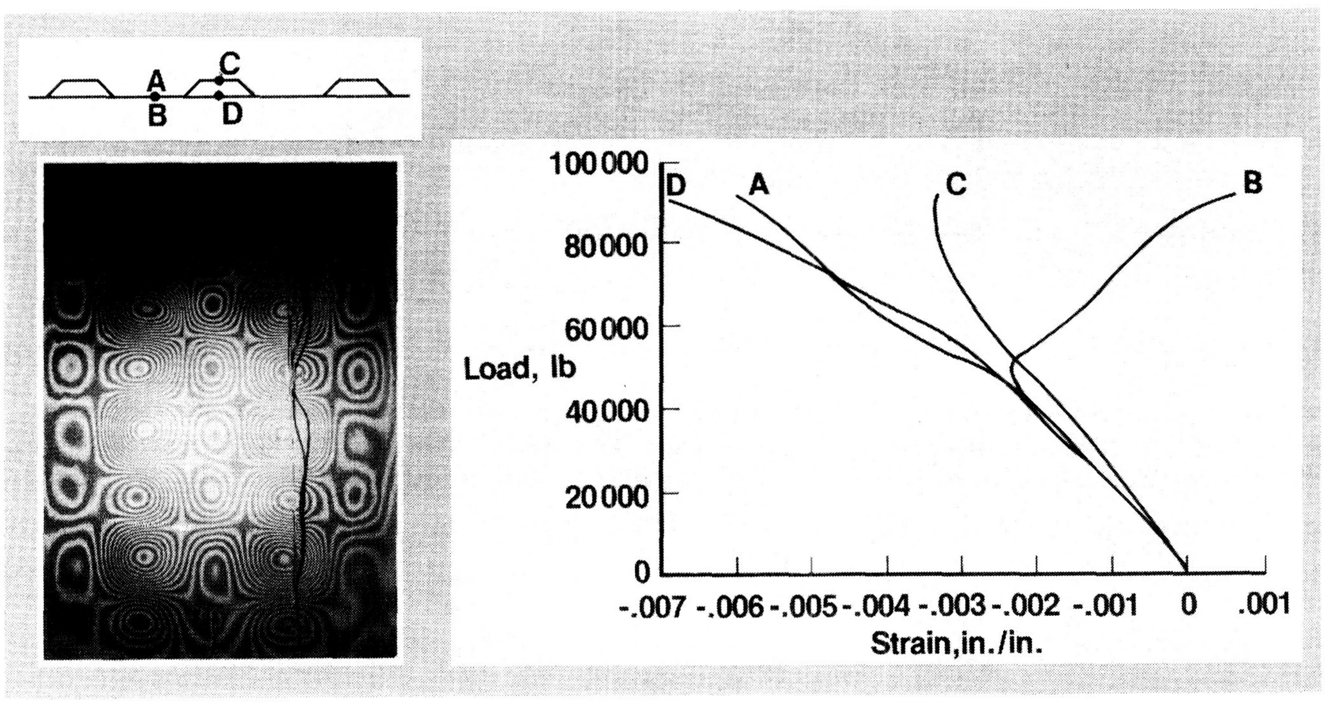


Figure 4

TYPICAL COVER PANEL FAILURE MODE

The typical failure mode for the cover panel specimens is shown in figure 5. On the left and center are front and rear views of the failed cover panel. The failure propagated across the bottom bay of the cover panel from one edge of the specimen to the other. The front of the specimen is painted white to provide a reflective surface for the moire fringe technique. The close-up photograph on the right shows the failure around the center stiffener. The hat stiffener has separated from the skin as indicated by the darker gap region slightly to the right of the scale. The center stiffener separated from the skin due to large post-buckling deformations, and the resulting damage propagated to fail the specimen.

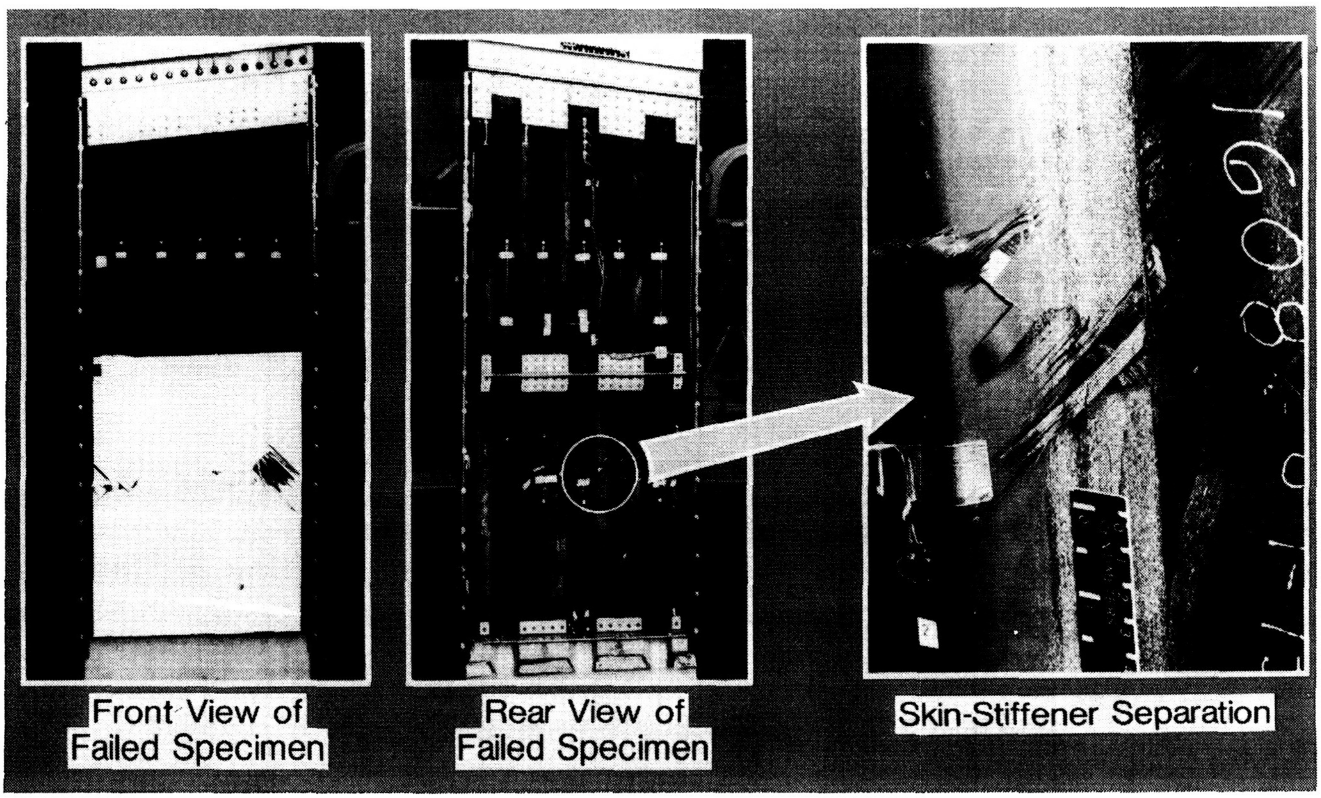


Figure 5

COMPARISON OF FAILURE LOADS FOR COVER PANEL SPECIMENS

Below is a comparison of the failure loads for the statically tested covers (control specimens) and the cover panels subjected to durability testing (conditioned specimens). The first bar represents the failure loads of the ten control tests. The range of failure loads is from 88000 lb to 96000 lb with an average failure load of 93000 lb. The remaining bars show the failure loads for the conditioned cover panel specimens. (See fig. 6.) The results indicate that the six conditioned cover panel specimens failed within, or slightly above, the failure range of the control specimens. The high-strain cover panels, which were loaded to design ultimate load during one cycle of the durability testing, showed no strength reduction. These results indicate that environmental conditioning and cyclic loading did not affect the performance of these cover panel specimens.

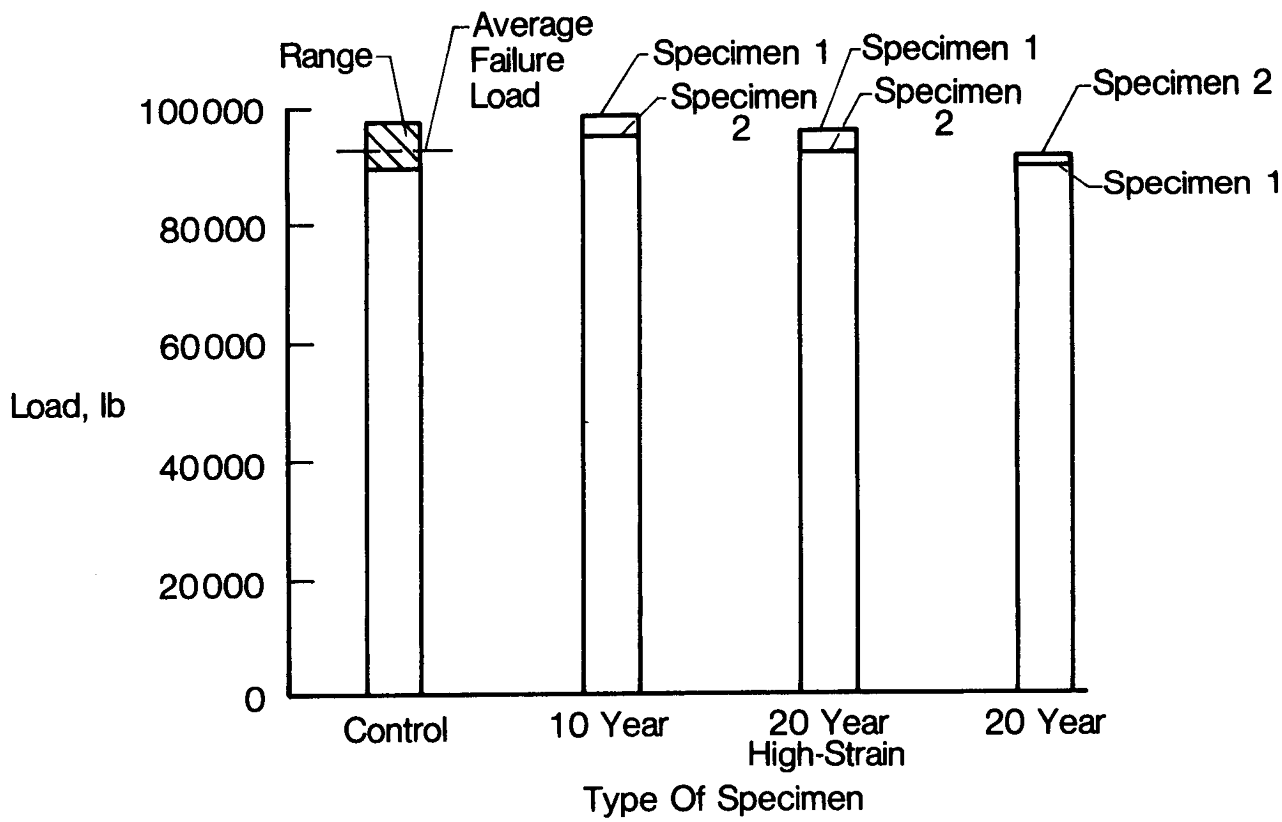
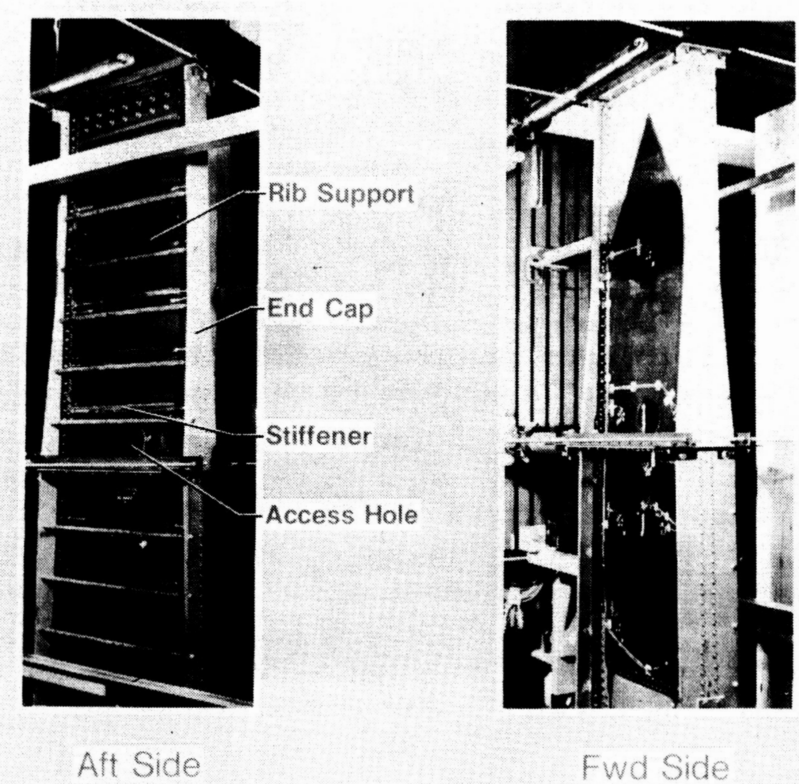


Figure 6

SPAR SPECIMENS

Two views of a typical leading-edge spar specimen (refs. 1, 3) are shown in figure 7. The spar specimen is 84 inches long, 28 inches wide at root and 21 inches wide at the tip. It is constructed from T300/5208 graphite-epoxy tape. The aft view of the spar shows several rib supports and stiffeners on the spar web. The spar specimens had two aluminum spar caps and three access holes. Some specimens had a fourth hole toward the tip. The forward view shows the smooth front surface of the specimen.

Lockheed constructed 22 of these spar specimens, ten of which were statically tested (control). The remaining 12 were subjected to long-term durability testing (conditioned). Five of the conditioned specimens were statically tested at NASA Langley Research Center to assess their residual strength. Two 10-year conditioned specimens and three 20-year specimens were tested at Langley. One of the 20-year conditioned specimens had sustained some damage around the edge of the bottom access hole prior to residual strength testing.



Specimen Conditions	Number Tested
Control (Calac)	10
10 Years of Conditioning	2
20 Years of Conditioning	3

Figure 7

TEST SET-UP FOR THE SPAR SPECIMENS

The test set-up for the front spar specimens is shown below. The test set-up consists of the specimen and four aluminum support frames which are attached to the laboratory floor. Eight threaded steel tie rods, attached to the frames, prevent lateral buckling at the spar specimen. The specimen is secured to the laboratory floor by use of a steel attachment base plate, and is loaded by two hydraulic jacks which bend the spar toward a steel back-stop. (See fig. 8.)

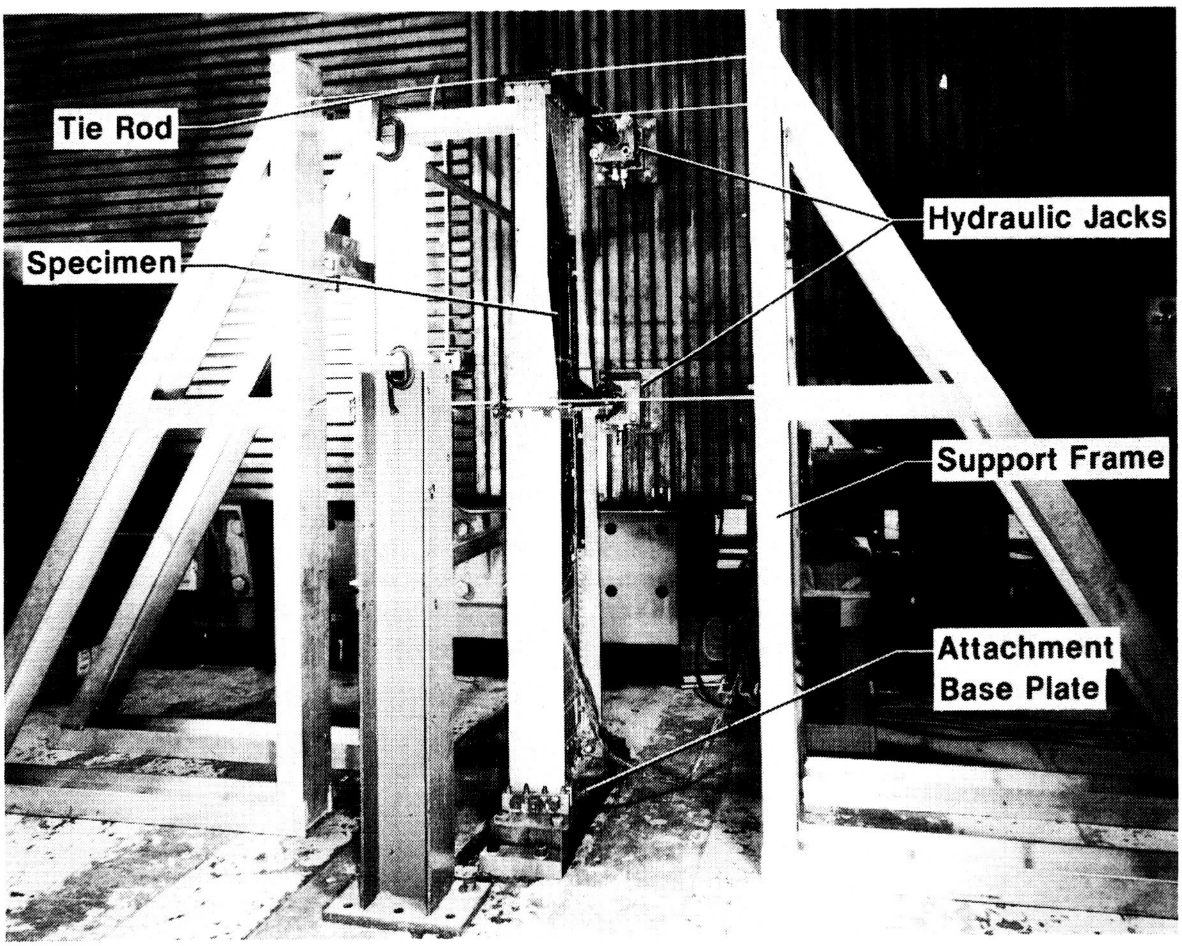


Figure 8

TYPICAL RESULTS FOR THE SPAR SPECIMENS

A photograph of a typical moire fringe pattern representing the buckling mode of the middle and lower access holes is shown in figure 9. A large amount of deformation around the upper edge of both access holes is indicated by the closely space fringes. The accompanying plots shows some typical strain gage results from a set of back-to-back strain gages located at the edge of the middle access hole.

Strain reversal occurs at an applied load of 25000 lb, indicating the onset of buckling. Buckling occurred well above the design limit load of 14000 lb. The specimen was loaded into the post-buckling range and failed at 32000 lb.

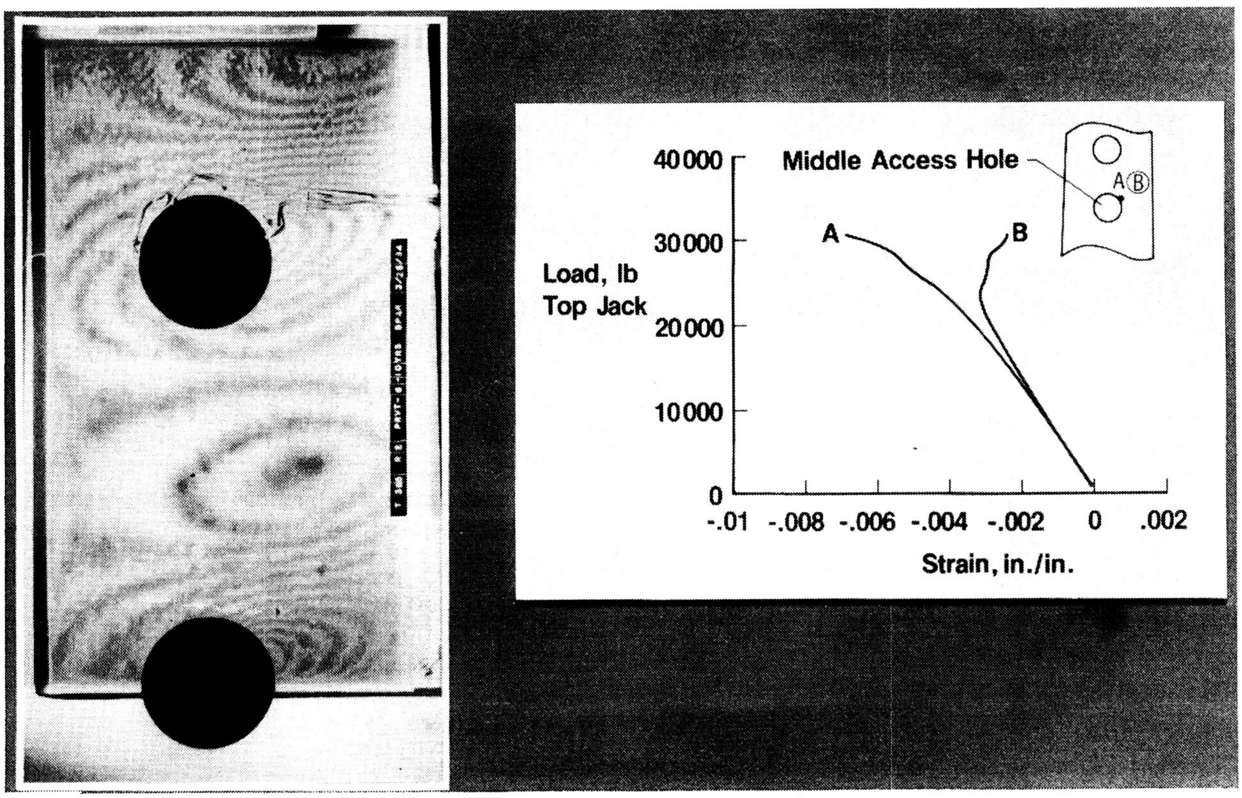


Figure 9

TYPICAL FAILURE MODE OF SPAR SPECIMENS

The typical failure mode of the spar specimens is shown in figure 10. The photograph on the left shows a shearing type failure which occurred around the middle access hole and propagated to both of the spar caps. As a secondary failure effect, the shear failure propagated along the spar caps causing fasteners to pull through the spar.

The photograph on the right is a close-up view of the failure and shows a large amount of delamination around the middle access hole. These local delaminations initiated at the edge of the middle access hole and propagated to fail the specimen. As the shear failure propagated across the spar web, two stiffeners located near the middle access hole separated from the spar web.

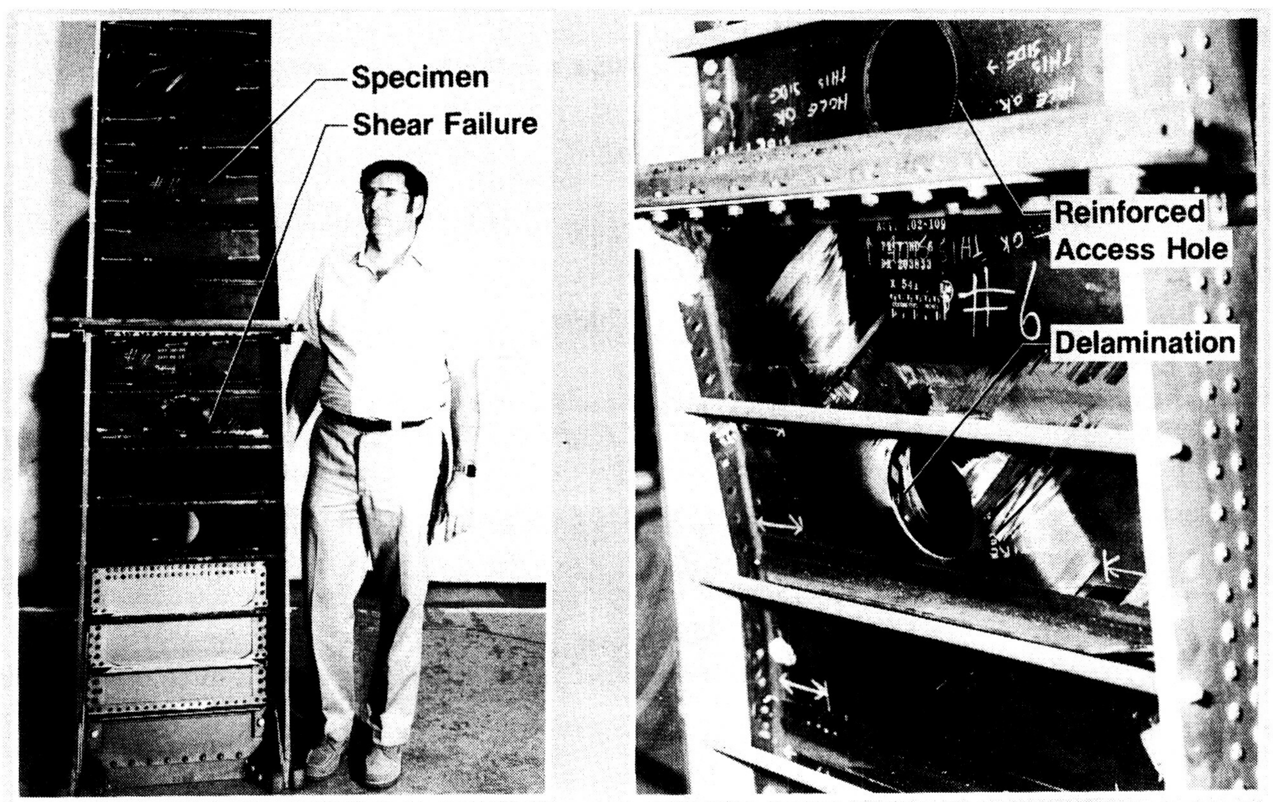


Figure 10

FAILURE MODE OF DAMAGED SPAR SPECIMEN

The failure mode of a 20-year conditioned spar specimen containing damage is shown in figure 11. Visual inspection of the spar specimen revealed, prior to residual-strength testing, a significant amount of delamination near the edge of the lower access hole. An ultrasonic inspection of the spar specimen identified an area of extensive damage around the lower access hole as indicated by the shaded region in the sketch on the left. A photograph of the failed specimen is shown on the right. The damaged spar failed differently from the other specimens, failing along the bottom access hole. However, the damaged spar failed within the failure load range of the other spar specimens. It appears that the damage affected the failure mode, but did not greatly affect the failure load of the damaged specimen.

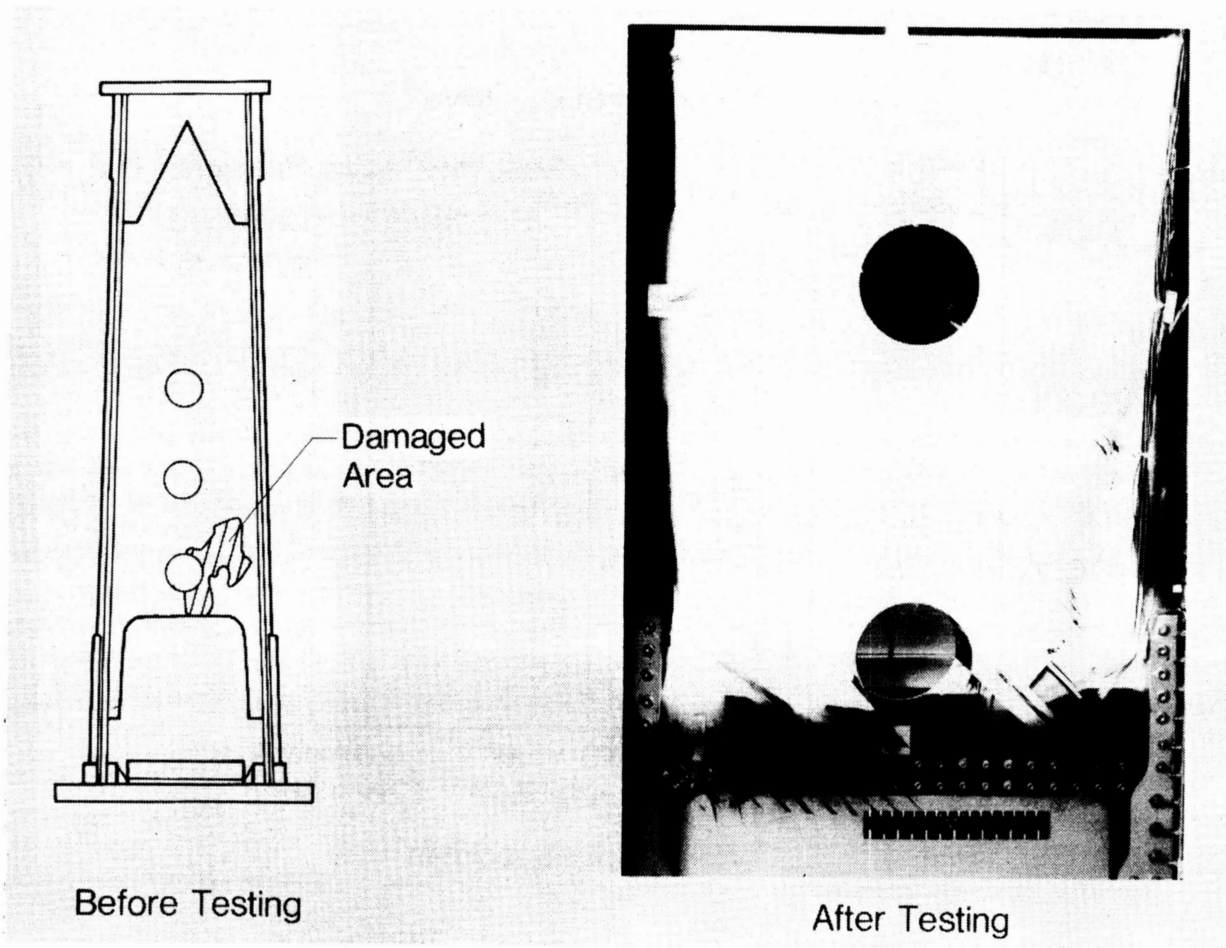


Figure 11

COMPARISON OF FAILURE LOADS FOR SPAR SPECIMENS

Plotted below are the failure loads for the statically tested spar (control) specimens and the spar specimens (conditioned) subjected to durability testing. The first graph represents the failure loads of ten control specimens. The failure loads range from 26000 lb to 31000 lb and the average failure load is 28000 lb. The remaining graphs show the failure loads of the conditioned spar specimens (fig. 12). The results indicate that all five conditioned specimens failed within the failure range of the control specimens or slightly above the average failure load for the control specimens. The spar containing damage failed at 31000 lb, which is above the average failure load for the control specimens. The results indicate that environmental conditioning and cyclic loading did not affect the failure loads of the conditioned spar specimens.

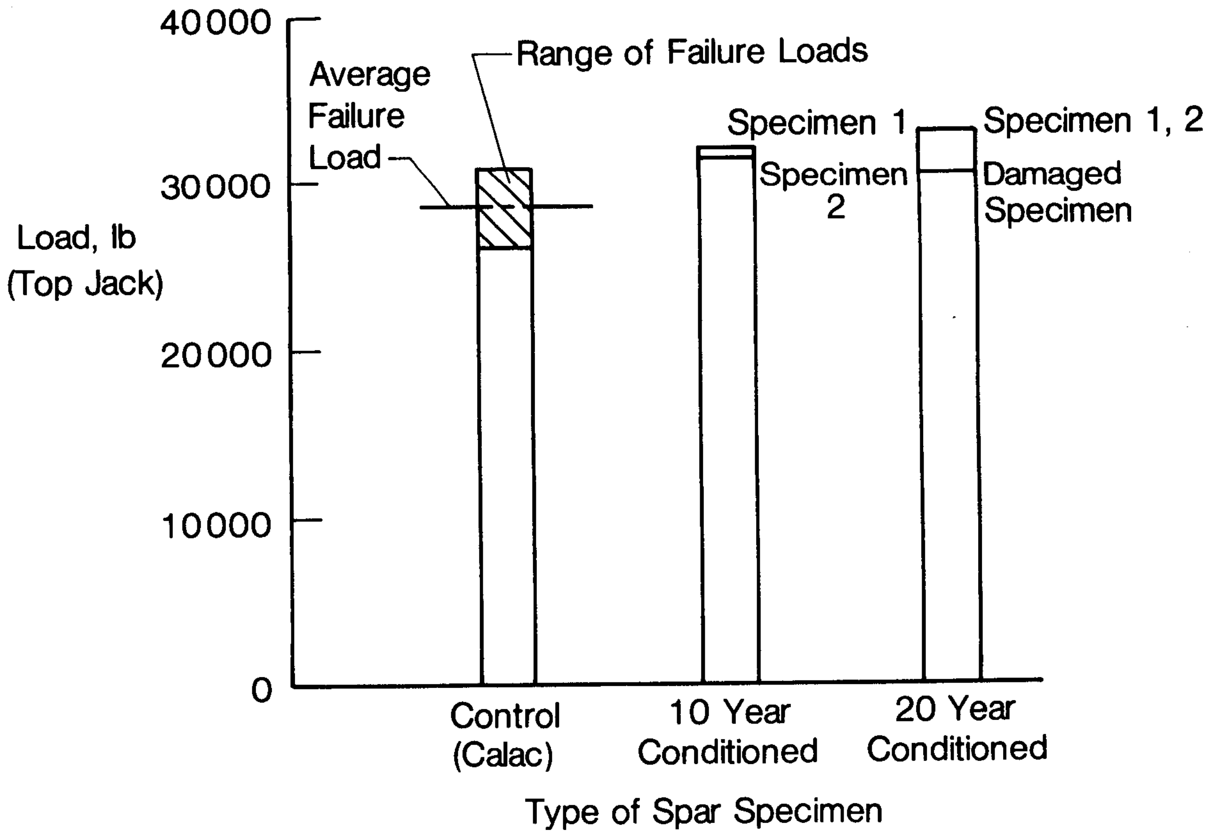


Figure 12

CONCLUDING REMARKS

- The failure loads of the specimens tested in this study were unaffected by simulated flight service conditioning.
- The cover panel specimens failed following large postbuckling deformations.
- The high-strain cover panel specimens did not experience additional strength reduction.
- The undamaged spar specimens failed around the middle access hole in the postbuckling range.
- The damaged spar specimen failed within the failure load range of the control specimens.

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

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