N95- 28470

TEST AND ANALYSIS RESULTS FOR COMPOSITE TRANSPORT FUSELAGE AND WING STRUCTURES

NASA
58-05
51348
P-25

Jerry W. Deaton
NASA Langley Research Center
Hampton, VA

Susan M. Kullerd
Lockheed Engineering & Sciences Company
Hampton, VA

Ram C. Madan and Victor L. Chen
Douglas Aircraft Company
Long Beach, CA

INTRODUCTION

Automated tow placement (ATP) and stitching of dry textile composite preforms followed by resin transfer molding (RTM) are being investigated by researchers at NASA Langley Research Center and Douglas Aircraft Company as cost-effective manufacturing processes for obtaining damage tolerant fuselage and wing structures for transport aircraft. The Douglas work is being performed under a NASA contract entitled "Innovative Composites Aircraft Primary Structures (ICAPS)." Data are presented in this paper to assess the damage tolerance of ATP and RTM fuselage elements with stitched-on stiffeners from compression tests of impacted three-J-stiffened panels and from stiffener pull-off tests. Data are also presented to assess the damage tolerance of RTM wing elements which had stitched skin and stiffeners from impacted single stiffener and three-blade-stiffened compression tests and stiffener pull-off tests.

The design concepts for stitched/RTM fuselage and wing panels were developed under previous NASA contracts and details are presented in references 1 and 2. The design criteria that the selected fuselage and wing concepts must satisfy are given in reference 3.

716
717

ICAPS TEST ARTICLES

The ICAPS test articles being evaluated are outlined in figure 1. The ATP fuselage elements include three-J-stiffened compression and J-stiffened pull-off specimens. The ATP crown panels were fabricated by Hercules, Inc., Magna, Utah, under contract to Douglas Aircraft Company using Hercules IM7/8551-7 graphite fiber reinforced toughened epoxy composite material. The skins and stiffeners were fabricated separately and cocured together. The skin and stiffener had the same layup $[0/90/45/0/-45/90]_S$. The RTM fuselage elements also include three-J-stiffened compression and J-stiffened pull-off specimens. The RTM crown panels had the same stacking sequence for the skin and stiffener as the ATP fuselage elements and utilized AS4 graphite uniweave fabric. Stiffeners were stitched to the skin and the assembly was pressure resin transfer molded with Shell 1895 epoxy resin using fixed volume tooling.

The damage tolerance of the fuselage elements was determined from impact tests performed on the compression and pull-off specimens. The impact energy for all fuselage elements was between 10 and 20 ft-lbs, which was the range of impact energy levels needed to obtain barely visible damage, and was accomplished by using either a 0.5-inch diameter or 1.0-inch diameter hemispherical drop weight impacter.

The RTM wing elements tested include single stiffener compression, three-blade-stiffened compression, and blade-stiffener pull-off specimens. The wing panels were fabricated from stitched skins and stiffeners utilizing AS4 graphite uniweave fabric. The skin has 54 plies with ply orientations of $[0/45/0/-45/90/-45/0/45/0]_{3S}$, and the stiffeners are 72 plies with the same layup as the skin. The stiffeners are stitched to the skin and then resin transfer molded with 3501-6 epoxy resin.

The damage tolerance of the wing elements was also determined from impact tests performed on wing element specimens. The impact energy for all impacted wing specimens was 100 ft-lbs, which is the cut off energy level for detectability, and was accomplished by using a 1-inch diameter hemispherical drop weight impacter.

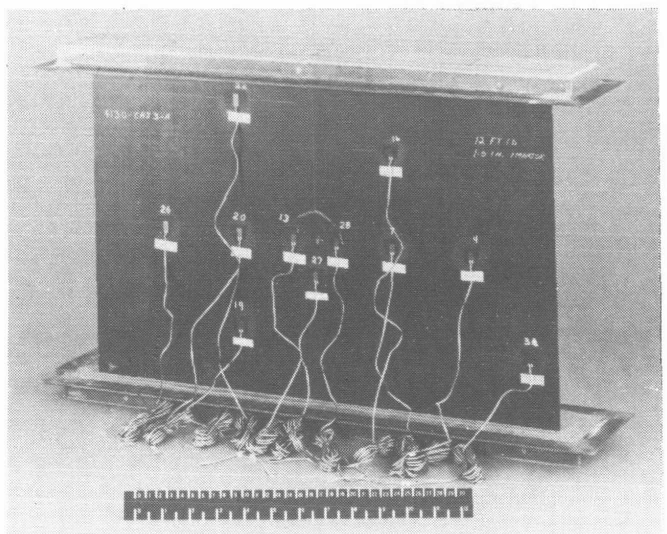
All RTM fuselage and wing elements were fabricated by Douglas Aircraft Company. All testing was performed at NASA Langley Research Center except as noted.

- **Auto tow placed (ATP) fuselage elements**
  - **IM7/8551-7**
  - **Three-J-stiffened compression tests**
  - **J-stiffened pull off tests**
- **Resin transfer molded (RTM) fuselage elements**
  - **AS4 Uniweave fabric/Shell 1895**
  - **Stitched stiffener**
  - **Three-J-stiffened compression tests**
  - **J-stiffened pull off tests**
- **Resin transfer molded wing elements**
  - **AS4 Uniweave fabric/3501-6**
  - **Stitched skin and stiffeners**
  - **Three blade stiffened compression tests**
  - **Single stiffener compression tests**
  - **Blade stiffened panel pull off tests**

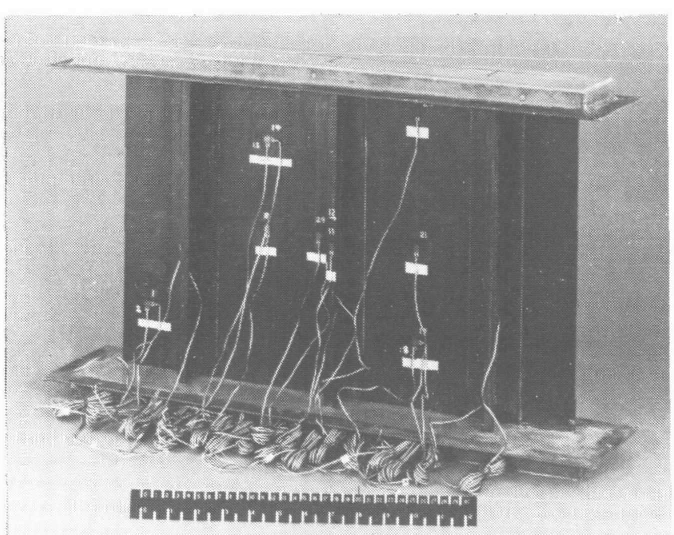
Figure 1

THREE-J-STIFFENED COMPRESSION PANEL

A typical fuselage compression panel is shown in figure 2. Specimens were nominally 21-inches wide and 15-inches long. The ends of each panel were potted using a room temperature potting compound. The ends of the panels were then machined flat, square, and parallel to each other. All impacted panels were impacted (prior to the potting procedure) from the skin side, midway between the specimen ends which were clamped during impact. Impact locations were either mid-bay, over the center stiffener, or at the flange edge of the center stiffener. Impact energy levels were selected which resulted in barely visible damage at each impact location. Each specimen was strain gaged as shown in figures 2a and 2b. The skin side of each compression panel was then spray painted white in order to use Moiré fringe interferometry to obtain buckling loads, mode shapes, and mode changes during the compression tests.



(a) Skin side



(b) Stiffener side

Figure 2

TEST SETUP FOR J-STIFFENED COMPRESSION TESTS

A 300 kip hydraulic test machine was used to apply compression loads to the specimens; see figure 3. In addition to the strain gages, seven LVDT's were used to monitor specimen displacements. One was used to monitor overall specimen shortening and two were used on each stiffener, one for out-of-plane displacements and one for stiffener rolling. Most specimens were tested at 0.02 in./min and strains and displacements were recorded continuously using an IBM PC-based data acquisition system.

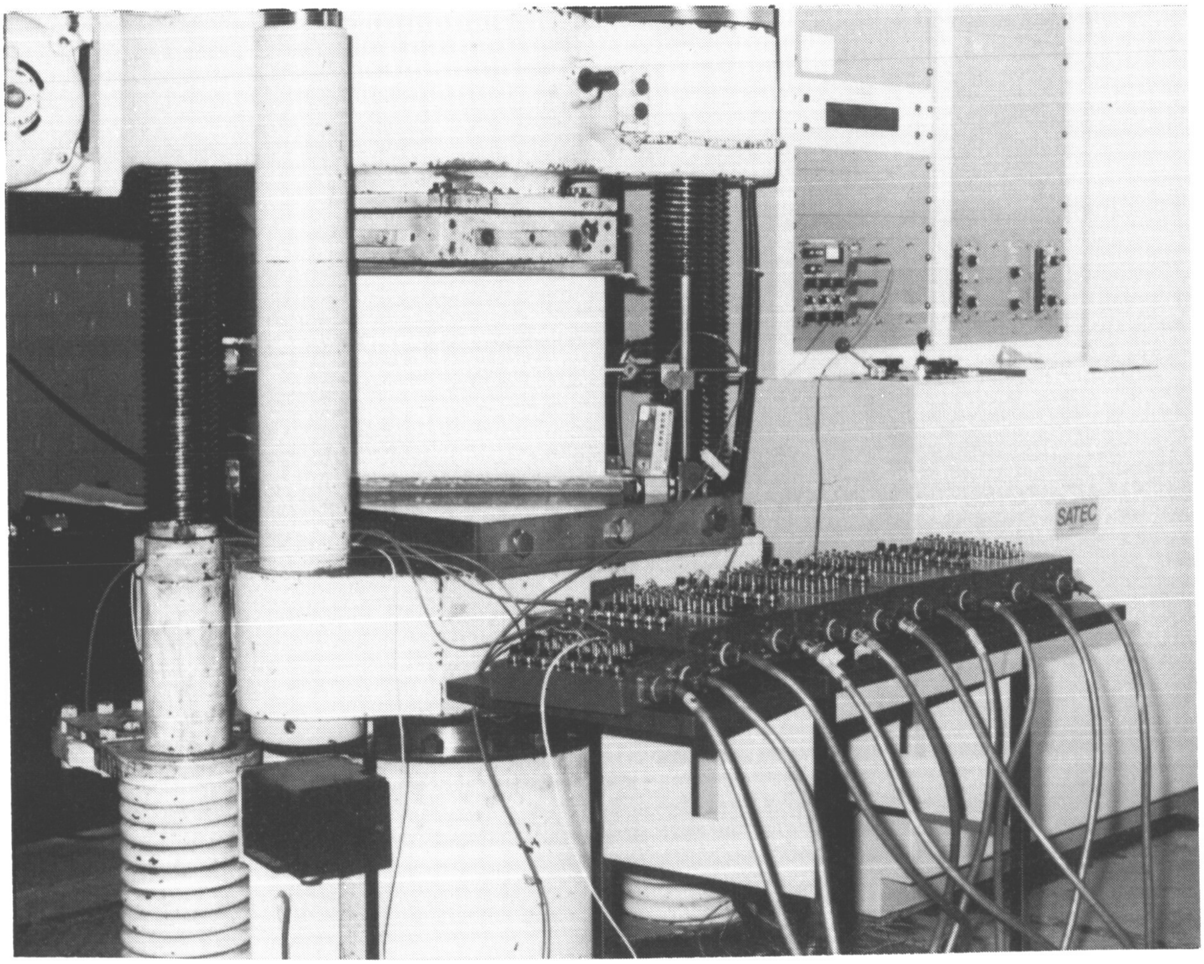
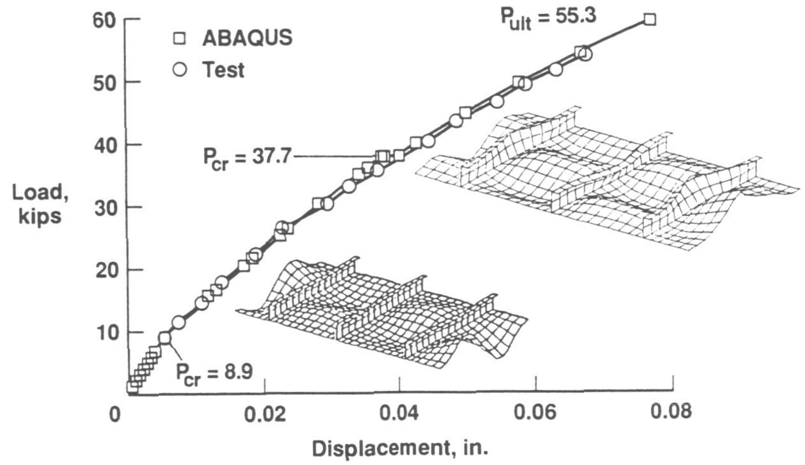
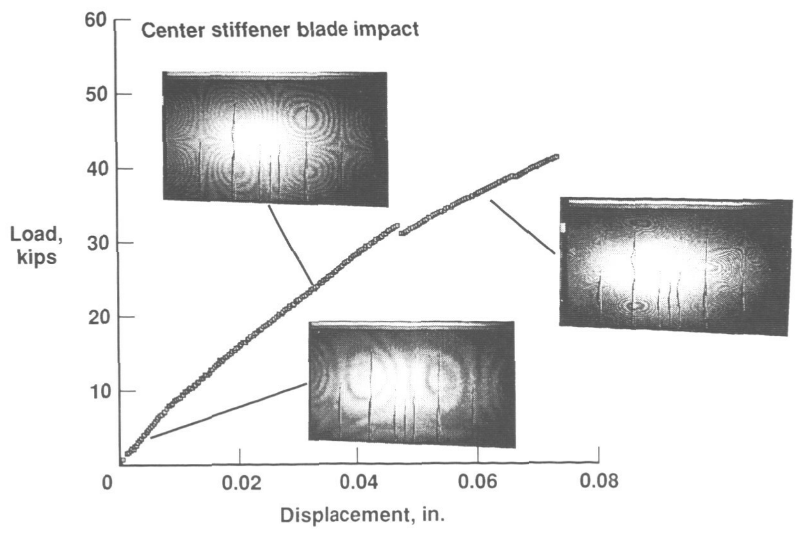


Figure 3

Figure 4 shows typical load-shortening data plots for the ATP and stitched/RTM fuselage J-stiffened panels under compression. Also shown in figure 4a are finite element simulation results obtained for the ATP panel using the ABAQUS code. Numerical predictions are shown to agree very well with test results. The ATP panel behaved linearly up to about 9 kips, then buckled into a one-half-wave mode (not shown) due to wide free edges. This mode smoothly grew into a two-half-wave mode slightly above the first P_{cr}, but the FEA showed lack of convergence and a nonlinear buckling (bifurcation) procedure was adopted. A three-half-wave mode was found numerically at 37-38 kips while tests showed this mode occurred around 35 kips with a loud popping sound. The FEA provided convergent results beyond 60 kips but predicted crippled stiffeners at about 55 kips, which is the measured failure load. Finite element simulation for the stitched/RTM fuselage panel has not been conducted because all necessary property data for the AS4 uniweave fabric/Shell 1895 material is not yet available. However, the load-shortening curve along with the Moiré fringe photographs, figure 4b, indicates that the stitched/RTM fuselage panel behaved similarly to the ATP panel under compression loading.



(a) ATP

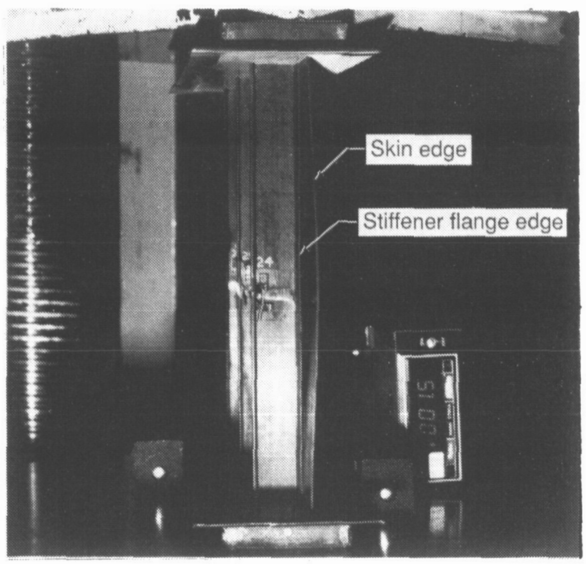


(b) RTM

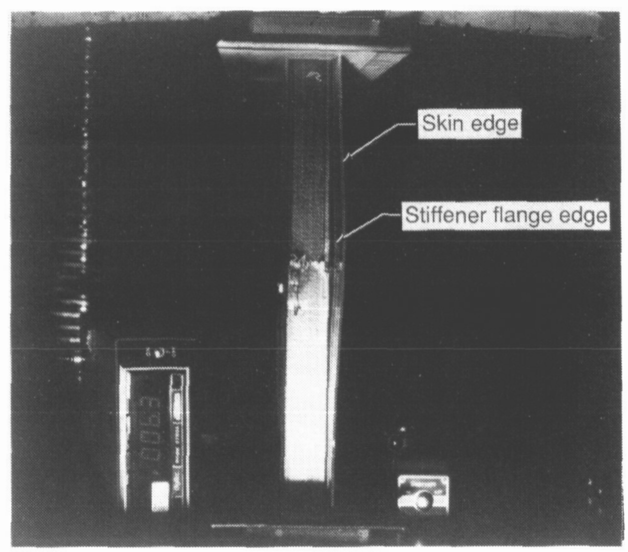
Figure 4

J-STIFFENED COMPRESSION PANEL FAILURE - SIDE VIEW

Figure 5 shows panel failure viewed from the side of the compression specimens. Figure 5a is for the ATP fuselage panel and illustrates stiffener crippling and skin/stiffener separation. Note that the stiffener flange edge is essentially straight whereas the skin edge is curved, which indicates that the buckled skin has separated from the stiffener. The load meter shown indicates a 1.5 kip load, which was applied to the specimen to better show the skin/stiffener separation. Figure 5b is for the stitched/RTM fuselage panel and illustrates a stiffener crippling failure but no skin/stiffener separation (both stiffener flange and skin are buckled) even with a 6.3 kip load applied as indicated by the load meter. All impacted stitched/RTM fuselage panel failures resulted in crippling failure of one stiffener without any skin/stiffener separation. Most impacted ATP panel failures involved crippling failures of all three stiffeners with accompanying skin/stiffener separation along with some degree of skin failure.



(a) ATP



(b) RTM

Figure 5

J-STIFFENED COMPRESSION PANEL FAILURES - STIFFENER SIDE

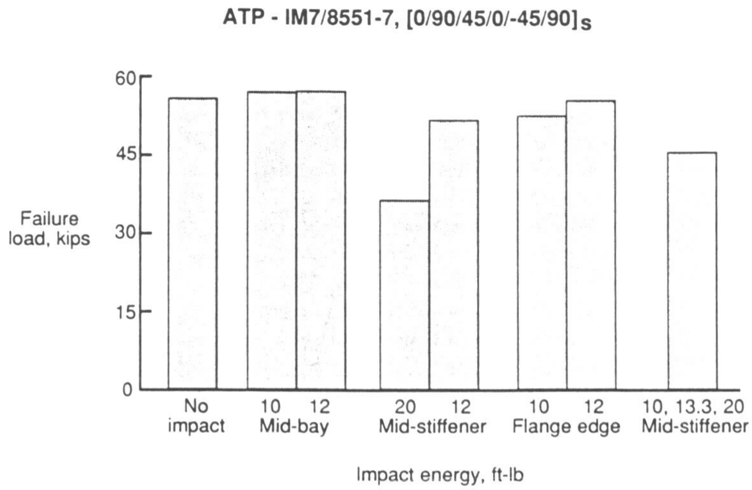
Additional photographs were obtained from the stiffener side of the failed panels. Figure 6a shows the same failed ATP fuselage panel shown in figure 5a, again, with a compression load of 1.5 kips applied. Crippling failure of all three stiffeners is evident along with some outer ply skin failure. Figure 6b shows the failed stitched/RTM fuselage panel shown in figure 5b with a 6.3 kip load applied. The stiffener on the right has failed in crippling, but the flanges remain attached to the skin. The failures shown in figures 5 and 6 are typical for all impacted J-stiffened compression panels.

(a) ATP

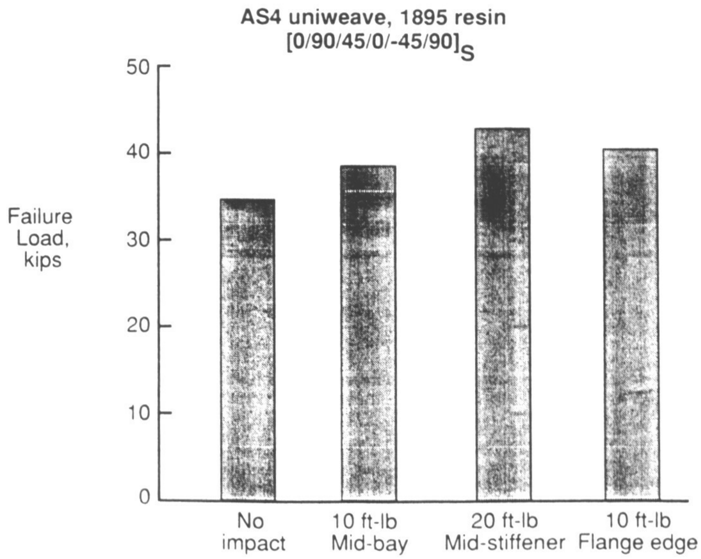
(b) RTM

Figure 6

The failure load of each J-stiffened fuselage panel tested is shown plotted in figure 7 for each impact location and impact energy level evaluated. The ATP and stitched/RTM data are shown in figures 7a and 7b, respectively. The lowest failure load (denoted by asterisk) shown on each plot was obtained from tests which were conducted at a displacement rate 2.5 times faster than all other tests. At this loading rate, the panel may not have had sufficient time to redistribute loads between skin and stiffeners when the panel underwent buckling and mode shape changes, thus causing a premature failure. Design criteria (ref. 3) for the fuselage panels requires an ultimate compression loading of 1700 lb/in. or about 35 kips for the specimens being evaluated. All impacted specimens exceeded this requirement. The higher failure loads obtained for the impacted ATP fuselage panels compared to the stitched/RTM fuselage panels can be attributed to the higher strength fiber and toughened resin system used in their fabrication.



(a) ATP



(b) RTM

Figure 7

ATP FUSELAGE STIFFENER PULL-OFF SPECIMEN

Cabin design pressure differential for the baseline aircraft fuselage is 9.1 psi limit, ref. 3. This is associated with flight loads to provide a limit condition at which there should be no detrimental structural deformation. This can be interpreted conservatively as no initial stiffener separation in the pull-off case due to pressure alone. Figure 8 shows an ATP fuselage stiffener pull-off specimen which was used to assess the effects of impact on stiffener pull-off load. Impacted pull-off specimens were machined from impacted three-J-stiffened panels as previously described for the compression tests. Pull-off specimens were 11.5-inches long and 4.5-inches wide and had 0.125-inch thick aluminum doublers bonded on each end on both sides with a room temperature curing adhesive. The bottom of the "J" stiffener was machined off to facilitate the introduction of pull-off loads into the specimen. Initial pull-off specimens utilized numerous strain gages (figure 8) to ensure that a uniform load distribution was obtained with the pull-off fixtures. After ensuring that the fixture was performing as desired, either 2 or 4 strain gages were used to aid in detecting the load at which initial skin/stiffener separation occurred.

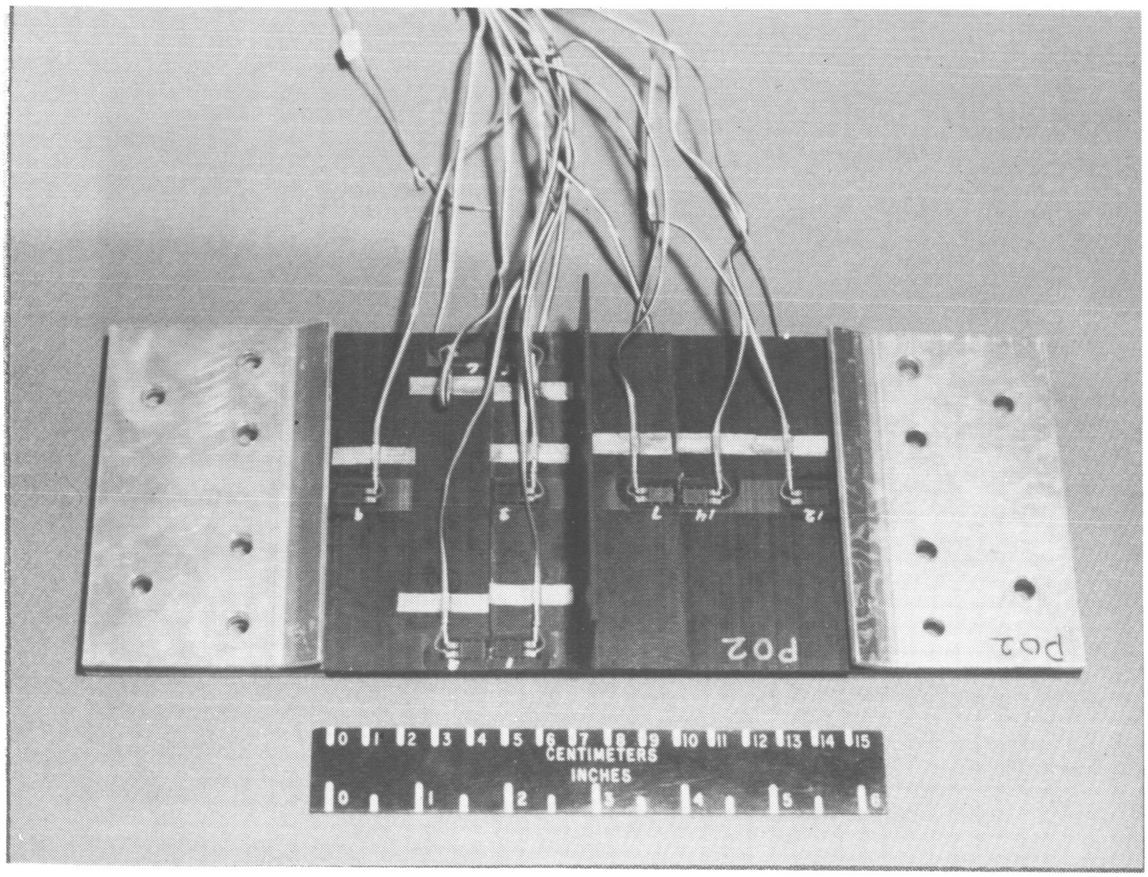


Figure 8

TEST SETUP FOR FUSELAGE STIFFENER PULL-OFF TEST

The setup for fuselage stiffener pull-off tests is shown in figure 9. The specimen was bolted to the loading fixture and 0.25-inch thick aluminum splice plates were bolted to the stiffener as shown in the figure. All bolts were torqued to a value of 60 in.-lb. and the assembly was placed inside an environmental chamber. Load was introduced into the loading fixture and splice plates through 0.75-inch diameter pins. Most tests were performed at room temperature; however, three ATP pull-off tests were performed at 180°F after the specimens were soaked in 160°F water for 13 days. For these three tests, strain gages were installed and then sealed by applying three layers of silicone waterproofing compound. All tests were performed at a displacement rate of 0.05 in./min and strain was recorded continuously throughout the tests. Photographs were taken during the tests to document the failure sequence.

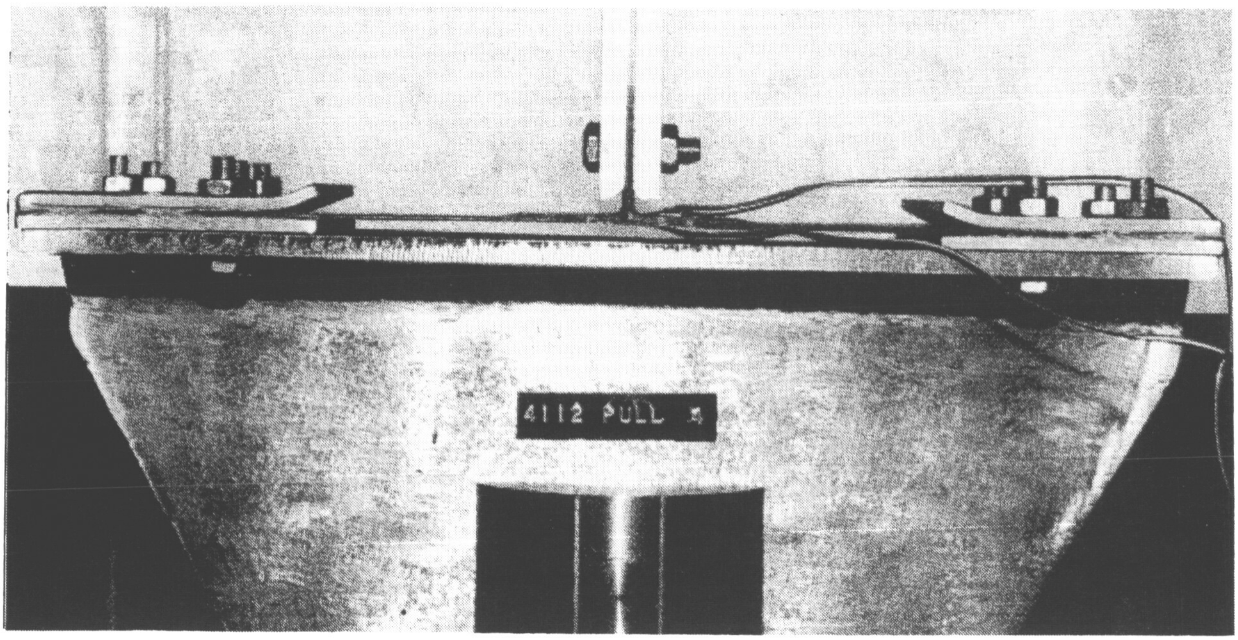


Figure 9

STITCHED/RTM FUSELAGE STIFFENER PULL-OFF TESTS

A typical load/strain plot for one of the stitched/RTM fuselage stiffener pull-off tests is shown in figure 10. Only two gages were installed on this specimen; gage 1 was located on the stiffener flange next to the upright portion of the stiffener, and gage 2 was located on the skin side of the specimen directly beneath the center of the stiffener. Both gages were oriented in the long dimension of the specimen. Load was applied continuously until the stiffener separated from the skin for the ATP specimens or until all stitches failed in one of the flanges for the stitched/RTM specimens. Initial skin/stiffener separation load was determined from a loud popping sound, visually (door to chamber was open except for hot, wet pull-off test), or from strain gage data. Photographs at 600, 800, and 1000 pounds of applied load are shown in the figure to illustrate the typical failure sequence; note the increase in skin deflection and crack growth between skin and stiffener with increasing load.

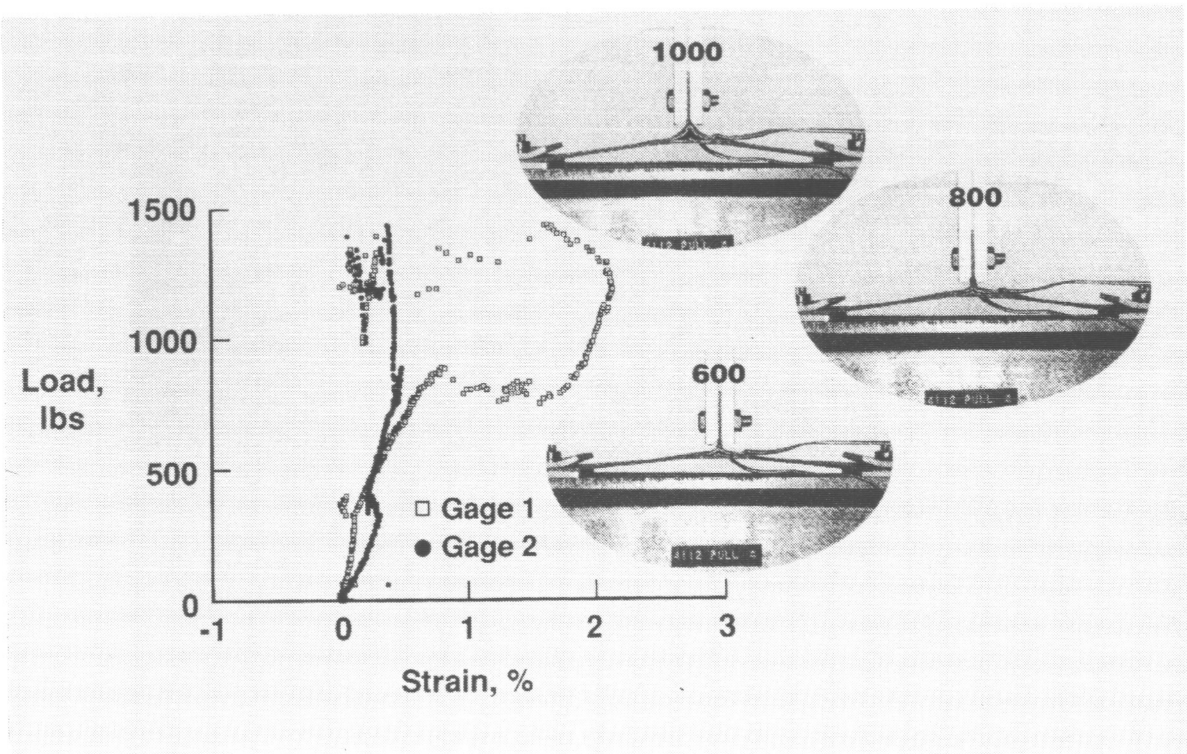


Figure 10

STITCHED/RTM FUSELAGE STIFFENER PULL-OFF FAILURE

Figure 11 shows a stitched/RTM fuselage stiffener pull-off specimen after failure. For this specimen, the photograph was taken with zero load indicated on the test machine. The skin deflection shown in the photograph was not permanent: the skin straightened out when it was removed from the loading fixture.

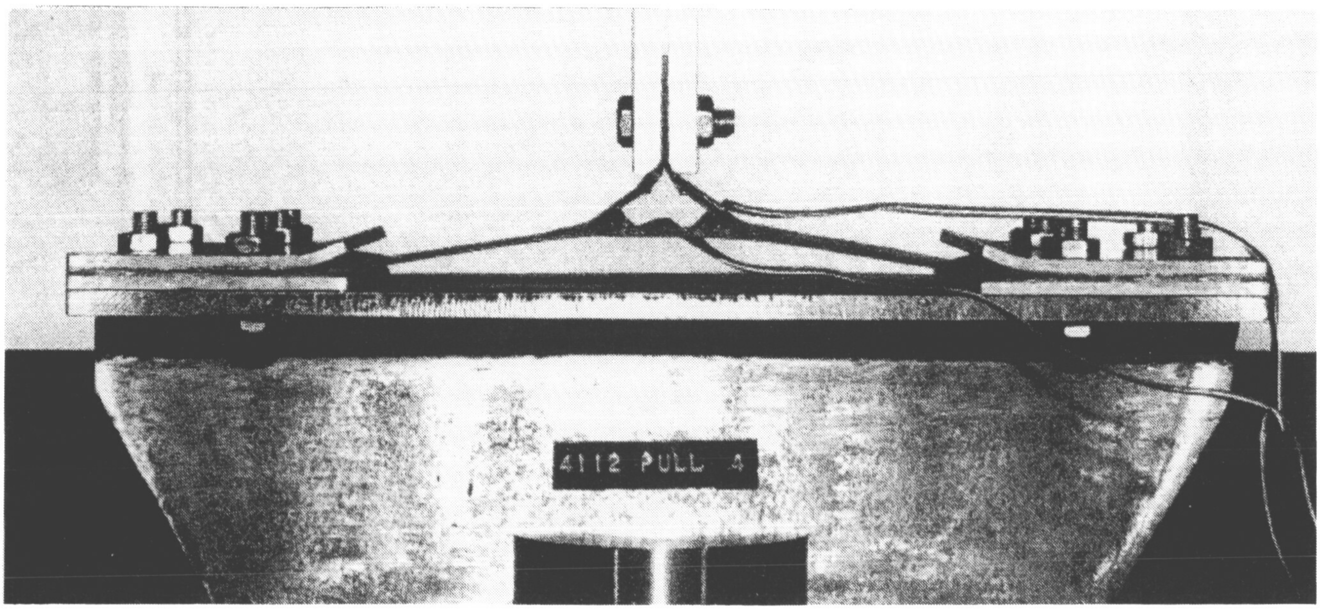


Figure 11

EFFECT OF IMPACT ON FUSELAGE STIFFENER PULL-OFF LOAD

Results obtained from the stiffener pull-off tests are shown in figure 12a and 12b for the ATP and stitched/RTM fuselage stiffener pull-off specimens, respectively. It should be noted that the ordinate shown for the ATP data is only one-half of that shown for the stitched/RTM fuselage pull-off data. The shaded bars correspond to the pull-off load at which skin/stiffener separation initiated and the open bars represent failure load. Results shown for the ATP specimens which were not impacted are the average of three tests at room temperature (RTD) and the average of three tests at 180°F after a 13-day water soak in 160°F water (HW). The data indicate that the ATP specimens subjected to the water soak and elevated temperature test conditions had reduced failure and skin/stiffener separation loads of about 20 and 40 percent of the RTD values, respectively. All other data shown in figure 12a and 12b represent individual test results. The data indicate that the flange edge impact for the ATP specimens is the critical impact location for both skin/stiffener separation and failure load where a reduction of about 80 percent occurs. For the stitched/RTM specimens, no reduction in pull-off load or initiation of skin/stiffener separation is indicated due to impact energy level or impact location. Superior stiffener-to-skin integrity is indicated for the stitched/RTM fuselage concept where twice the ATP strength is indicated without damage and ten times the ATP strength with flange edge impact damage.

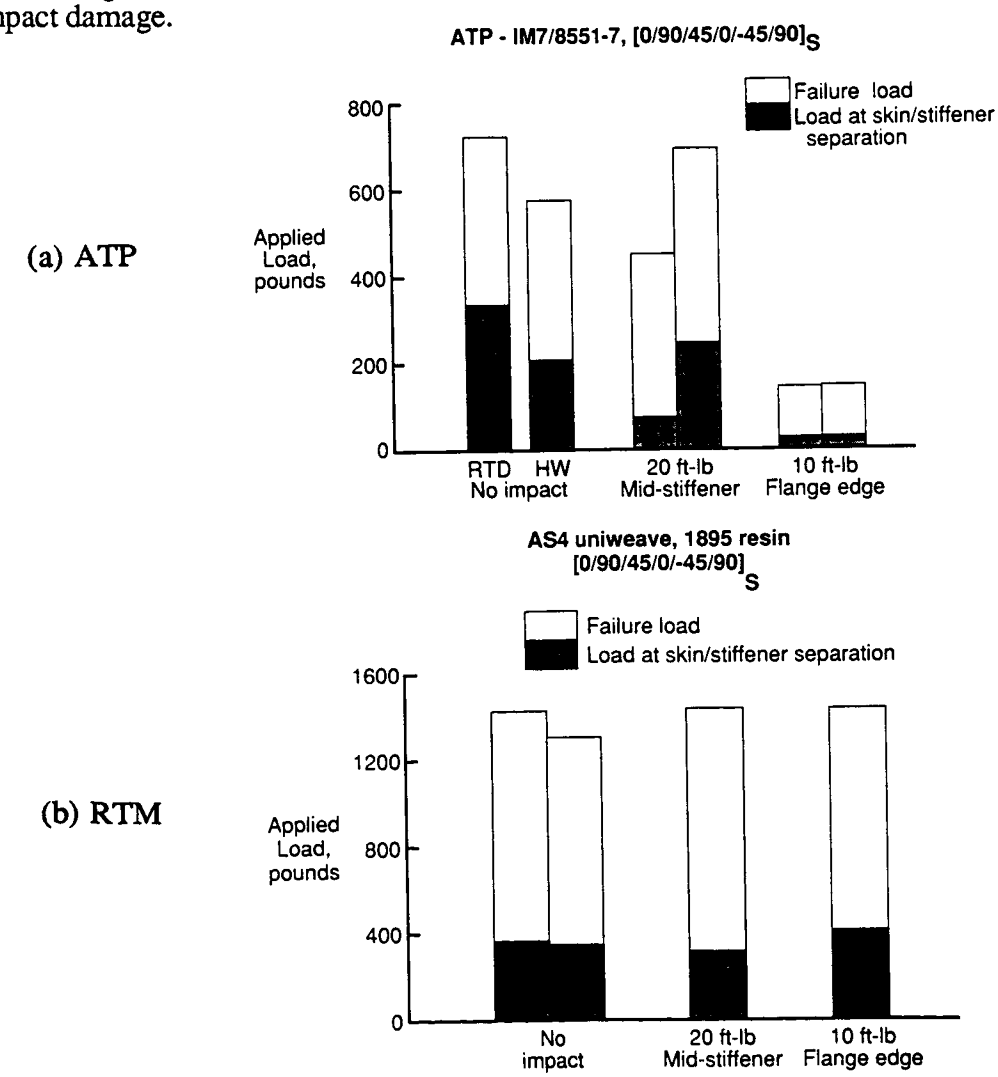


Figure 12

Figure 13 shows schematically the fabrication procedure for making preforms for wing panels. The 54-ply skin is made by stitching together six of the basic 9-ply subelements of AS4 uniweave fabric having the layup shown. The stiffener is made by stitching together eight of the 9-ply subelements to form the blade. Flanges are formed by folding out 4 subelements on each side and cutting them at different lengths to provide taper. A filler of prepreg tape is placed in the flange-to-blade joint and the flanges are then stitched to the skin. Additional information on this concept is detailed in the paper by S. Kullerd and M. Dow, titled "Development of Stitched/ RTM Composite Primary Structures," also presented at this conference. The preform is placed in a tool and resin transfer molded with 3501-6 epoxy resin.

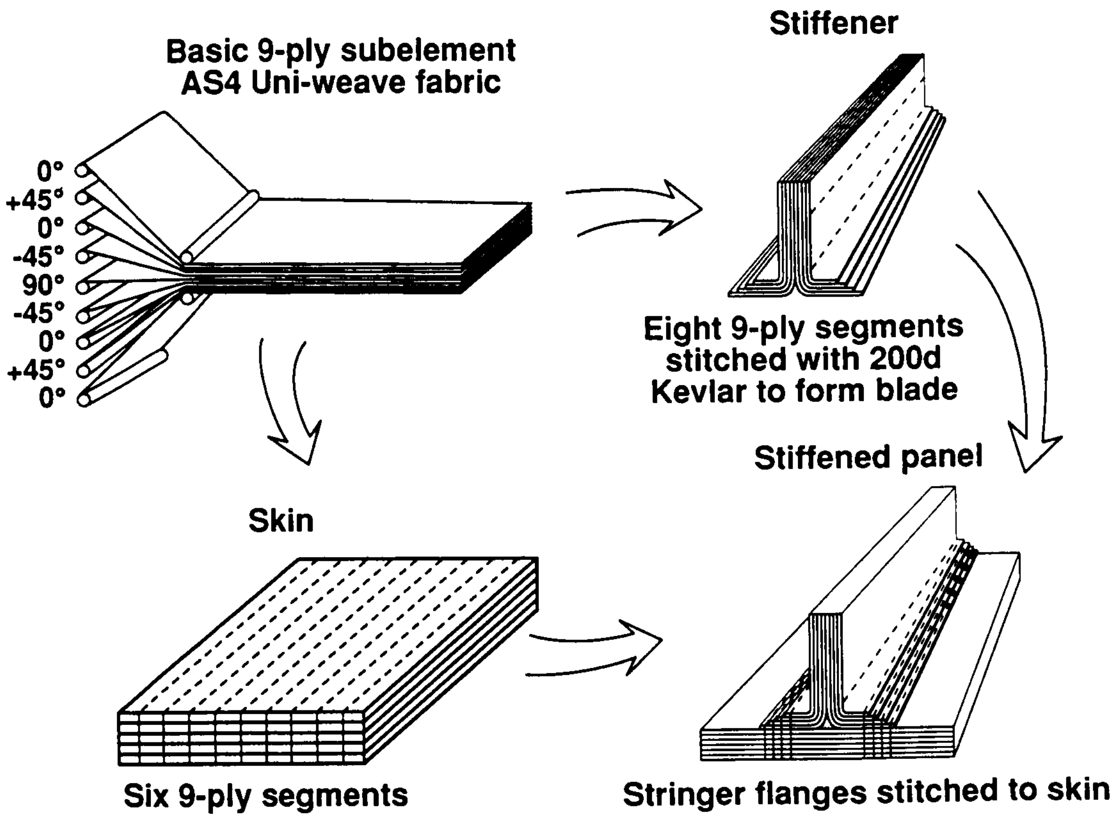


Figure 13

DAMAGE TOLERANT COMPRESSION PANELS

Compression panels used to assess the damage tolerance of stitched/RTM composite primary wing structures are shown schematically in figure 14. Six single-stringer specimens were machined from one 3-stringer panel after impact. Impacts were made on the skin side directly beneath a stringer or at the flange edge of a stringer in such a way that the impact location for single-stringer specimens was at the center length during compression testing. The impact energy for all impacted specimens was 100 ft-lbs, which is the cut off energy level for detectability, and was accomplished by using a 1-inch diameter hemispherical drop weight impacter. The panel was C-scanned before machining the single-stringer test specimens. Each single-stringer specimen was instrumented with three pairs of back to back strain gages.

The skin side impact locations for the three-stringer panels include mid-bay, mid-stringer, and flange-edge of the middle stringer. The ends of each three-stringer panel was supported along its width and clamped to a table during impact. Each end of the three-stringer compression specimens was potted in a room temperature potting compound. The ends were then machined flat, square and parallel to each other. Each three-stringer panel was instrumented with 15 strain gages and included back-to-back pairs on both the skin and center stringer. The compression tests were performed at a displacement rate of 0.05 in./min by Douglas Aircraft Company or their subcontractor.

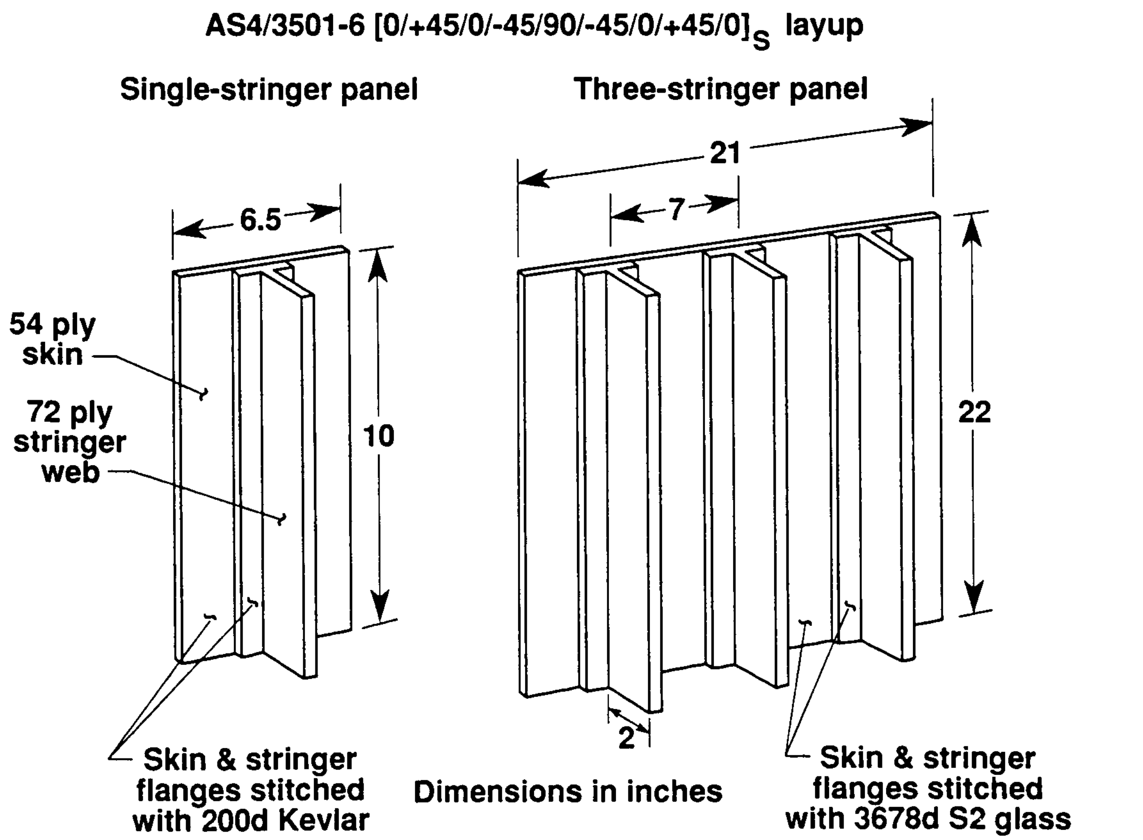


Figure 14

POST-IMPACT CRIPPLING STRENGTH OF SINGLE-STIFFENER WING ELEMENTS

The effect of a 100 ft-lb impact on the crippling strength of single-stiffener compression specimens is shown in figure 15. The shaded bars are the average obtained from two specimens. The flange-edge impacted specimens failed at a lower load than unimpacted and mid-stiffener impacted specimens. However, the reduction was less than 10 percent for individual specimens. All single-stiffener compression tests were performed by Douglas Aircraft Company.

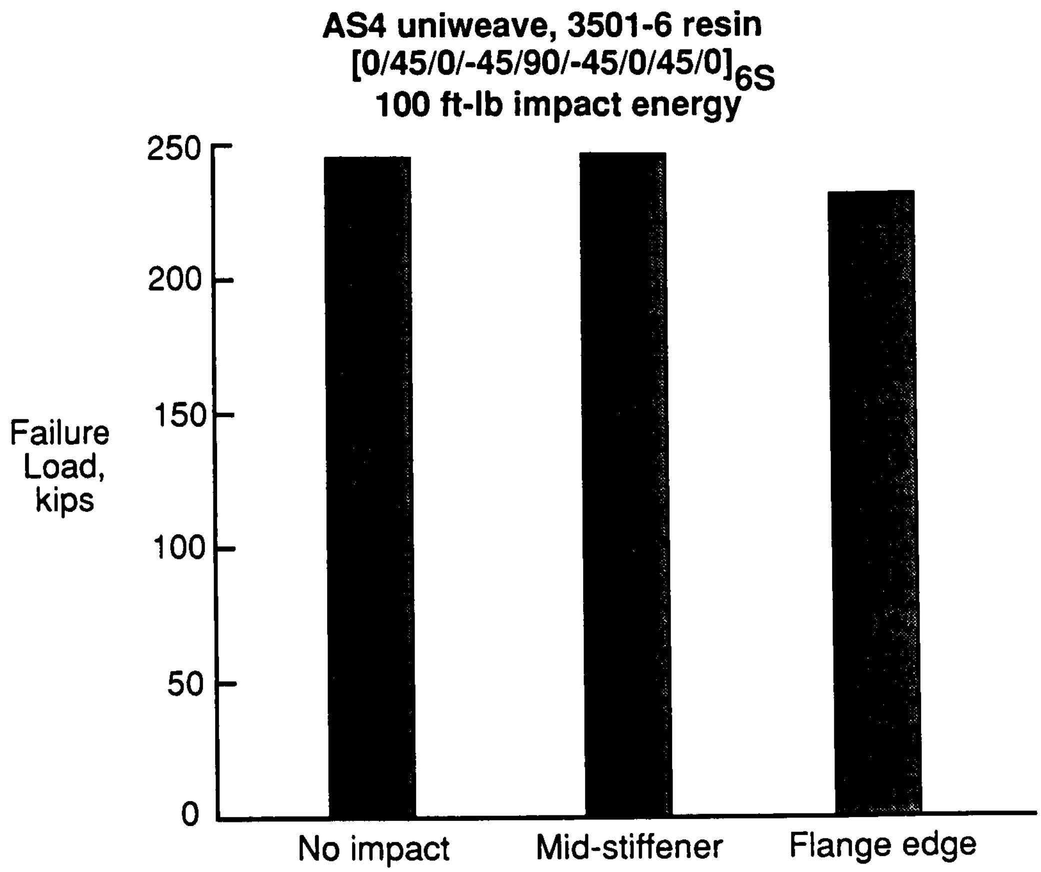
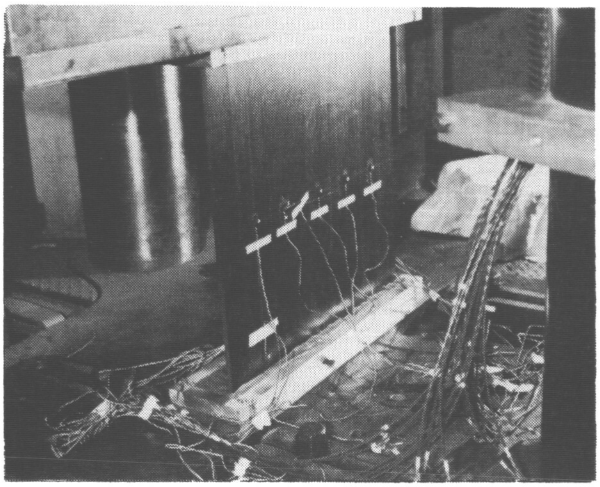


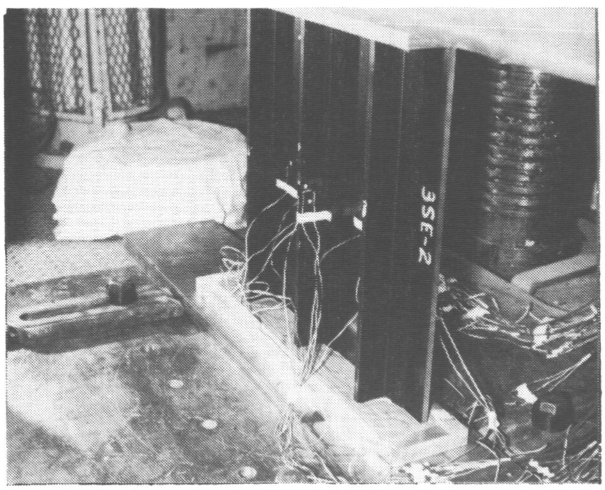
Figure 15

TEST SETUP FOR COMPRESSION TEST OF 3-STIFFENER WING PANEL

The test setup for the 3-stiffener wing panel is shown in figure 16. The panel shown (figure 16a) has been impacted at the mid-bay location. Figure 16b shows the same panel as viewed from the stiffener side. The panel was tested in DAC's 1.1-million pound capacity test machine at a displacement rate of 0.05 in./min. Three additional panels were tested at Hercules' Magna, Utah, test facility using their 1.5-million pound capacity MTS machine. All panels failed without any skin/stiffener separation and a slight bending (buckling) was observed just before panel failure.



(a) Skin side

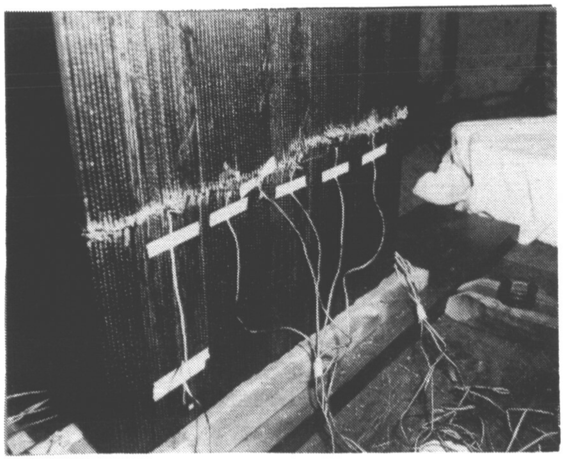


(b) Stiffener side

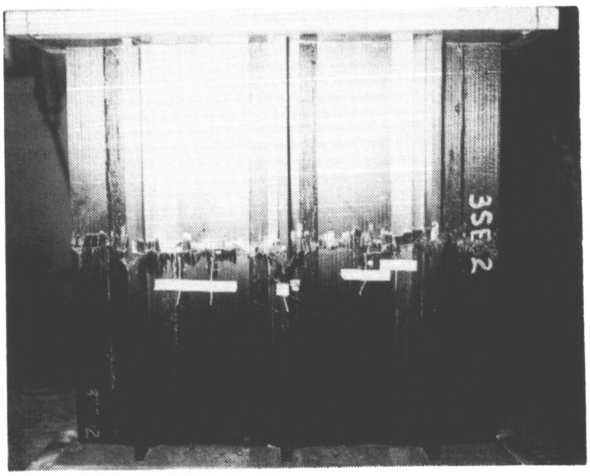
Figure 16

3-STIFFENER WING PANEL COMPRESSION FAILURE

Photographs of the mid-bay panel failure are shown from the skin side and stiffener side in figures 17a and 17b, respectively. The skin (figure 17a) failed through the impact location and all three stiffeners (figure 17b) failed. The stitching pattern used to fabricate the skin and stiffeners can be seen in these figures.



(a) Skin side



(b) Stiffener side

Figure 17

POST-IMPACT COMPRESSION STRENGTH OF 3-STIFFENER WING ELEMENTS

The failure load of each blade-stiffened wing panel tested is shown plotted in figure 18. Data are shown for panels impacted at mid-bay, mid-stiffener, and center stiffener flange-edge at an impact energy of 100 ft-lbs and are compared to the failure load of a panel without impact damage. Design criteria (reference 3) for the wing panels requires an ultimate compression loading of 23.6 kips/in. or about 496 kips for the 21-inch wide specimens being evaluated. All impacted specimens exceeded this requirement. The data indicate that the mid-bay impact is the most critical location for stitched panels subjected to compression loading where a reduction of about 20 percent in the failure load is indicated compared to the specimen which was not impacted. The panels impacted at the mid-stiffener and stiffener flange-edge at the 100 ft-lb energy level did not experience a reduction in load capability compared to the panel which was not impacted. The results shown in figure 18 are very encouraging when compared to the results obtained in reference 2 for mid-bay impacted panels fabricated from 1808I/IM6, a very damage tolerant material. The referenced panel was also 21-inches wide and had a 54 ply skin and 72 ply stiffeners of the same ply orientation as the stitched/RTM wing panel. The mid-bay impacted 1808I/IM6 panel of reference 2 failed at a load of 363 kips and the failure sequence consisted of skin/stiffener separation, skin buckling, and catastrophic failure.

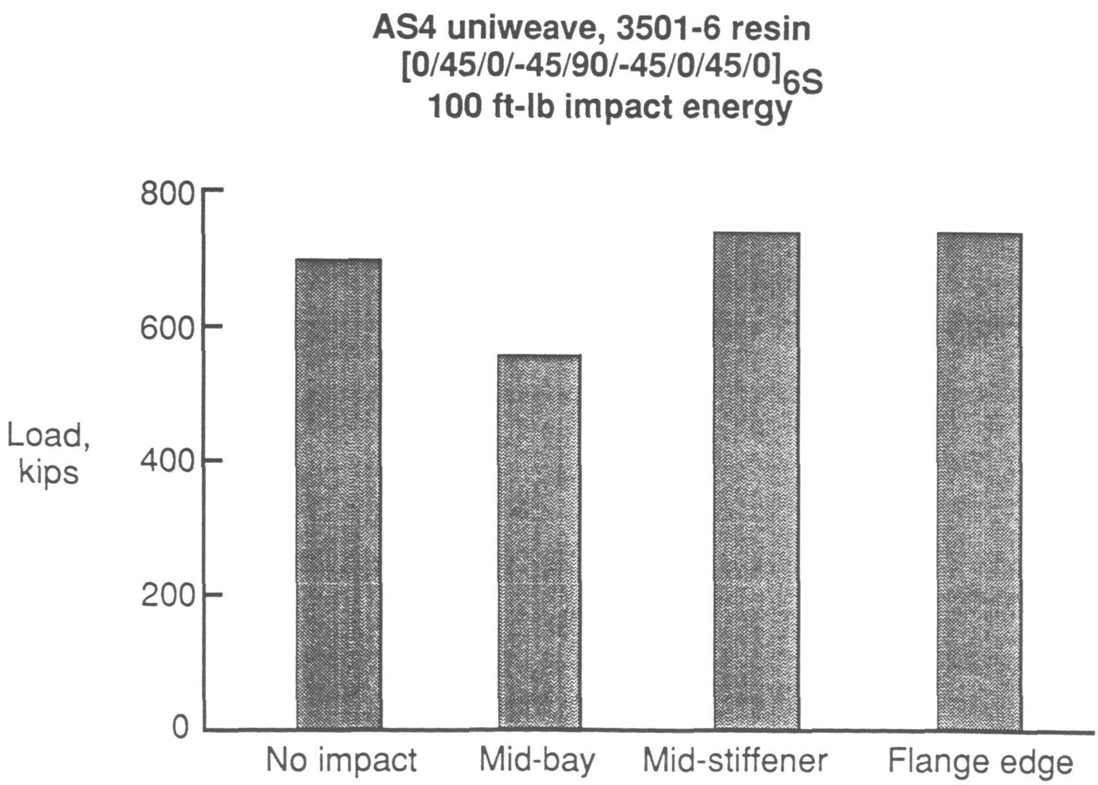


Figure 18

STITCHED/RTM WING STIFFENER PULL-OFF SPECIMEN

The ultimate design load for wing fuel tankage corresponds to the highest fuel pressure which is projected to occur during a 9g crash. Although the only criterion for this situation is not to rupture the tank, it is desirable not to experience total separation of skin and stiffener. The highest fuel pressure combined with the stiffener spacing results in a 327 lb/in. ultimate pull-off loading. Figure 19 shows a stitched/RTM wing stiffener pull-off specimen which was used to assess the effects of 100 ft-lb impacts on stiffener pull-off load. Pull-off specimens were machined from impacted 3-blade-stiffened panels as previously described for the compression tests. Wing stiffener pull-off specimens were 10.5-inches long and 4.5-inches wide. Each end of the specimen had a 0.125-inch and 0.5-inch thick aluminum doubler bonded to the bottom and top of the specimen, respectively. Wing pull-off specimens were instrumented with either 2 or 4 strain gages to aid in detecting the load at which initial skin/stiffener separation occurred.

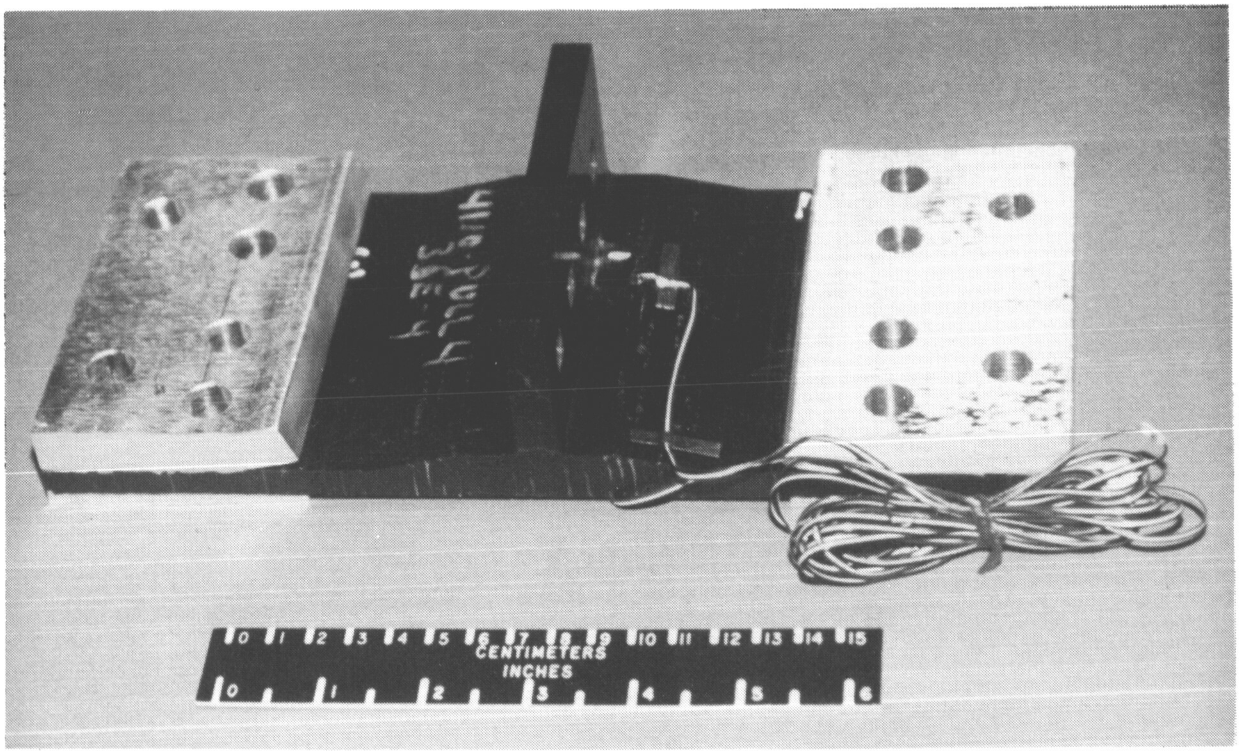


Figure 19

TEST SETUP FOR WING STIFFENER PULL-OFF TEST

The set up for wing stiffener pull-off tests is shown in figure 20. The setup is similar to that used for the fuselage pull-off tests; however, the loading fixture and splice plates were much thicker. The specimen was bolted to the loading fixture and 0.5-inch thick steel splice plates were bolted to the stiffener with 0.5-inch diameter bolts as shown in the figure. All bolts were torqued to a value of 75 ft-lbs and the assembly was pinned to the loading rods inside the environmental chamber. All wing pull-off tests were performed at room temperature at a displacement rate of 0.05 in./min and strain was recorded continuously throughout the tests. Photographs were taken during each test to document the failure sequence.

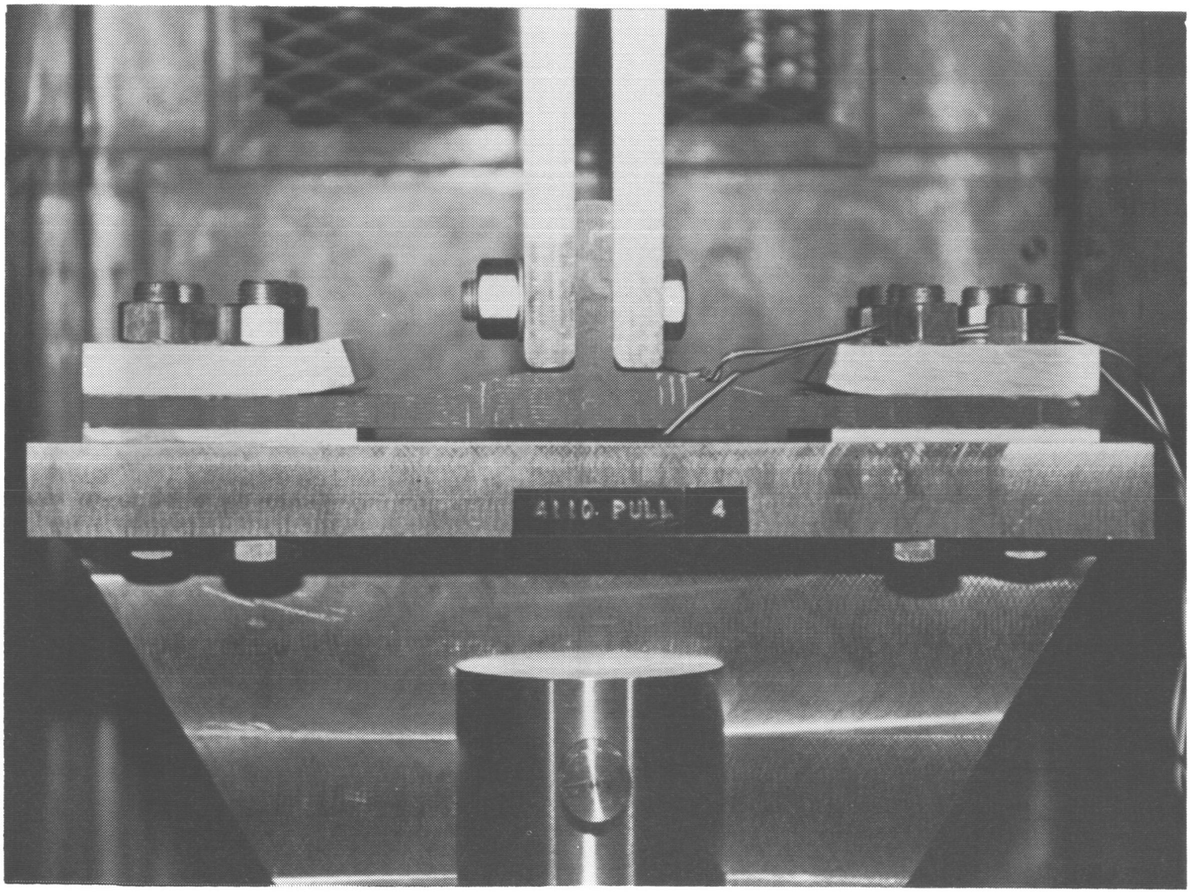


Figure 20

STITCHED/RTM WING STIFFENER PULL-OFF TESTS

A typical load/strain plot for one of the stitched/RTM wing stiffener pull-off tests is shown in figure 21. For the data shown, gage 1 was located on the stiffener flange and gage 2 was located on the skin side of the specimen directly beneath the center of the stiffener. Both gages were oriented perpendicular to the blade stiffener. Load was applied continuously until failure of all stitching on one of the flanges. Initial skin/stiffener separation load was determined visually, audibly, or from strain gage data. Photographs taken at 4000, 6000, and 8000 pounds of applied load are shown in the figure to illustrate the failure sequence at loads corresponding to failure of a line of stitching through the flange.

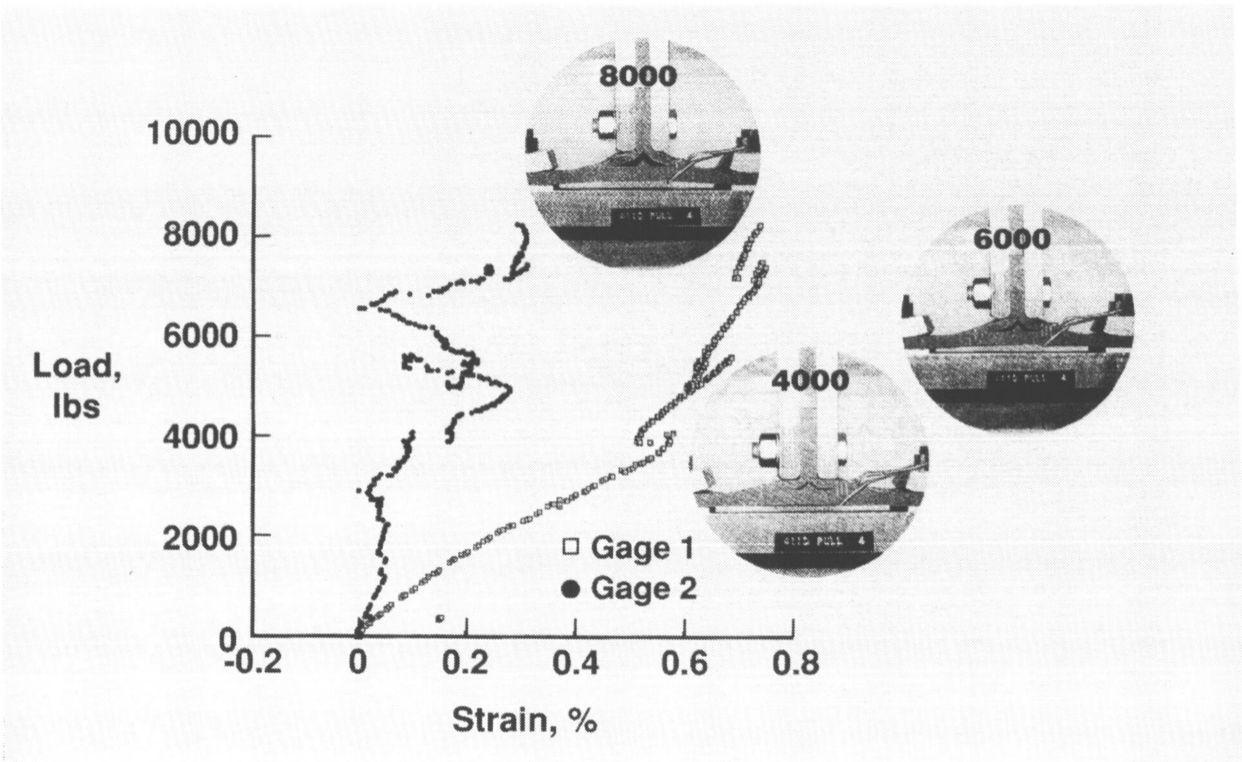


Figure 21

STITCHED/RTM WING STIFFENER PULL-OFF FAILURE

Failure of a stitched/RTM wing stiffener pull-off specimen is shown in figure 22. Failure consists of stitching breakage in each flange along with delamination between the 9-ply subelements of AS4 uniweave fabric. Note that the skin has returned to the straight preloading condition. A total of six wing pull-off specimens were tested and the failure shown in figure 22 is typical for both non-impacted and impacted specimens.

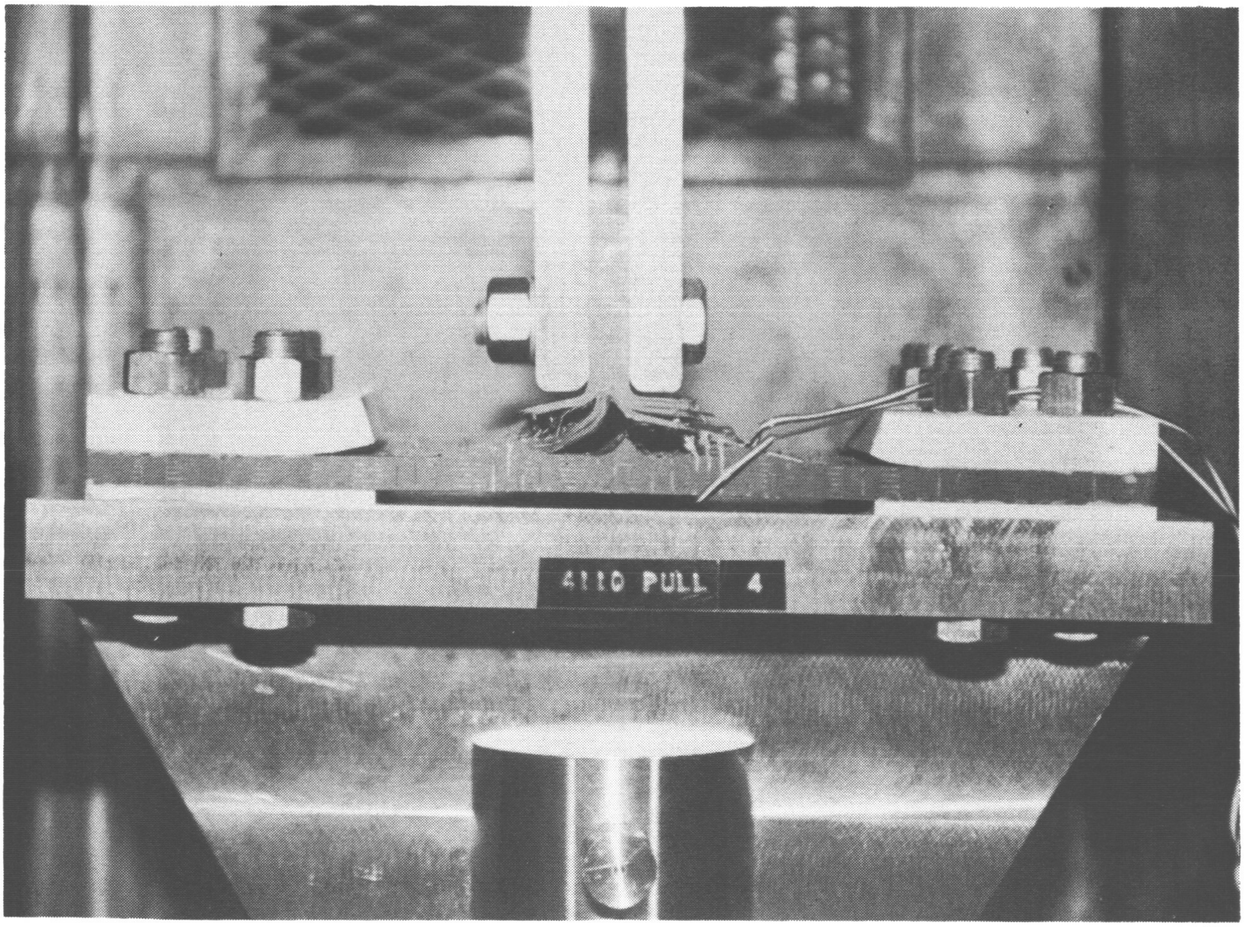


Figure 22

EFFECT OF IMPACT ON WING STIFFENER PULL-OFF LOAD

Results obtained from the wing stiffener pull-off tests are shown in figure 23. The shaded bars correspond to the pull-off load at which skin/stiffener separation initiated and the open bars represent the maximum failure load. Each bar represents an individual test. Recall that the ultimate pull-off load associated with the highest fuel pressure in the wing was 327 lb/in. which corresponds to approximately 1500 pounds of applied load for the 4.5-inch wide wing stiffener pull-off specimens. The data indicate that all specimens exceeded the ultimate load requirement without experiencing initial skin/stiffener separation. The failure load data indicate that the flange-edge is the critical impact location for this test where a reduction of approximately 37 percent in the pull-off load is noted. However, failure load exceeded the design ultimate requirement by a factor of three.

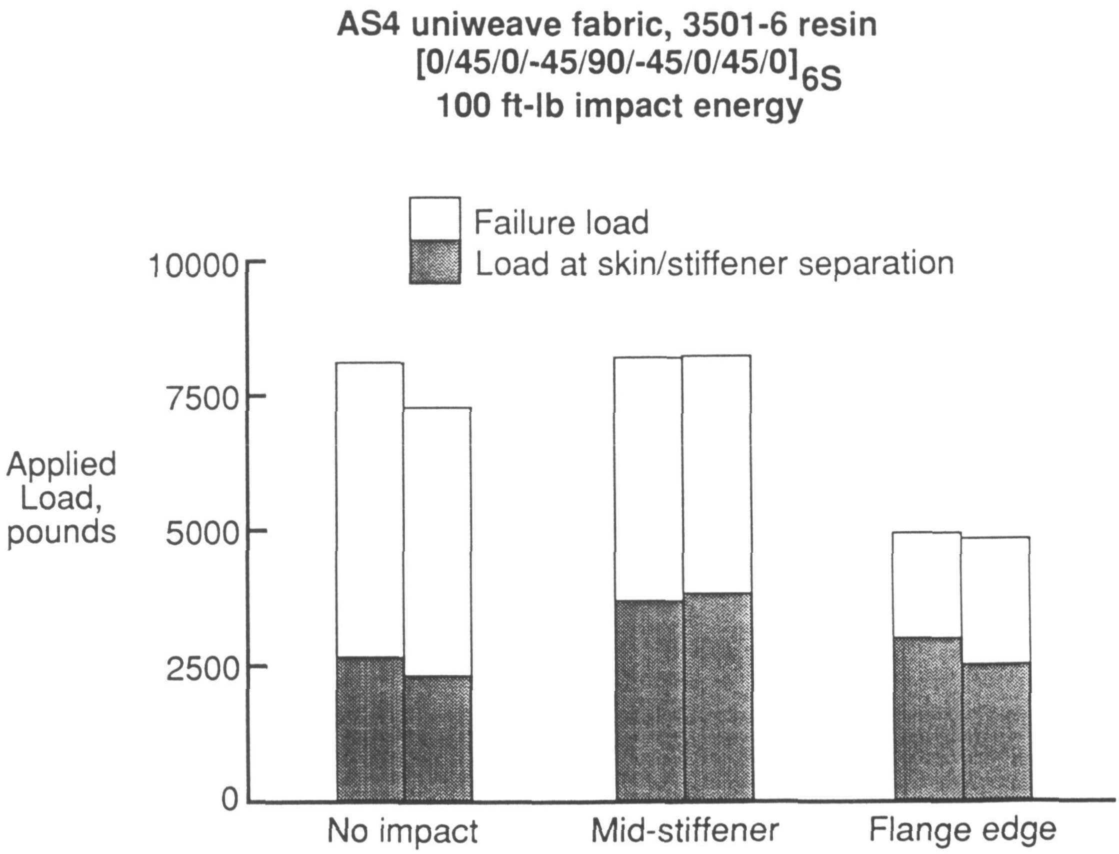


Figure 23

CONCLUSIONS

The damage tolerance of automated tow placement (ATP) and resin transfer molded (RTM) fuselage elements with stitched-on stiffeners has been determined from compression tests of impacted three-J-stiffened panels and from impacted stiffener pull-off tests. The damage tolerance of RTM wing elements which had stitched skin and stiffeners was also determined from impacted single-stiffener and three-blade-stiffened compression tests and impacted stiffener pull-off tests. The results of this investigation lead to the following conclusions:

- Fuselage Structural Elements
  - Both fuselage concepts met compression design goals with impact damage present.
  - Critical impact sites were identified for both ATP and stitched/RTM fuselage concepts: mid-stiffener for ATP and mid-bay for stitched/RTM compression tests.
  - Analysis correlated well with test results for ATP panel: predicted buckling and non-linear bifurcation load within 5 and 10 percent, respectively. Stitched/RTM laminate properties are being obtained for FEM analysis.
  - Stiffener pull-off failure load of ATP specimens were reduced 20 percent for hot-wet condition. No hot-wet pull-off test performed on stitched/RTM concept.
  - Superior stiffener-to-skin integrity for stitched stiffener fuselage concept demonstrated through pull-off tests: factor of 2 stronger than ATP without damage and factor of 10 stronger than ATP with damage.
- Wing Structural Elements
  - All three-stiffener and single-stiffener specimens met compression design goal after 100 ft-lb impact.
  - Mid-bay critical impact site for 100 ft-lb impact energy for three-stiffener compression panel test: 20 percent reduction in compression strength.
  - Flange edge critical impact site location for 100 ft-lb impact energy from stiffener pull-off tests where a 37 percent reduction in pull-off load was obtained. Failure load still exceeded the design requirement by a factor of three.
- The test results demonstrate that wing and fuselage structure meeting damage tolerance goals can be designed and fabricated using stitching and RTM processes.

REFERENCES

1. Sumida, P. T.; Madan, R. C.; and Hawley, A. V.: Test Results for Composite Specimens and Elements Containing Joints and Cutouts. NASA CR-178246, Aug. 1988.

2. Madan, R. C.: Composite Transport Wing Technology Development. NASA CR-178409, Feb. 1988.

3. Chen, Victor L., et. al.: Composite Technology for Transport Primary Structure. First NASA Advanced Composites Technology Conference, Seattle, WA, Oct. 29-Nov. 1, 1990, NASA CP-3104, Part 1, pp. 71-126.