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Study of Stitched and Unstitched Composite Panels Under Shear Loadings

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The airframe must be strong, rigid, and durable, yet as light in weight as safety will allow. To reduce structural weight, sometimes the aircraft structure is allowed to postbuckle in shear, a condition commonly known as diagonal tension. Postbuckled design will be used to meet the aggressive weight goal. This paper is focused on the comparison between stitched and unstitched panels under shear loadings designed to reduce the weight of aircraft components. Finite element analysis was used to determine the critical principal stresses, strains, and shear stresses. The results from the finite element analysis were compared with the actual testing data. It shows that stitching the stiffener flanges to the skin increases the shear strength capabilities of the test specimen by 43%. Moreover, a shear failure in the middle bay skin of the stitched specimen shows that the skin-stiffener delamination failure mode was eliminated.

Introduction

THE airframe must be strong, rigid, and durable, yet as light in weight as safety will allow. The lighter the aircraft, the more fuel efficient it will be. To reduce the weight of an aircraft, engineers have come up with different plans, such as using lighter materials and altering the structural design. Each change in the structural design will be carefully studied to ensure the aircraft integrity. This paper is focused on the study of stitched and unstitched shear panels designed to reduce the weight of an aircraft.

The skin on the AH-64 helicopter is loaded primarily in shear.[1] To reduce structural weight, the aluminum skin is allowed to postbuckle in shear, a condition known commonly as diagonal tension, and postbuckled design is used to meet the aggressive weight goal. The failure of diagonal tension panels can occur in several ways, other than exceeding the strength of the skin. Stiffeners surrounding the buckled skin can fail as a result of forced buckling when they are not stiff enough to resist the out-of-plane loads in the skin, and these same loads can pull the stiffener off the skin.[2,3] Stiffener sizing and attachment is very important in designing postbuckled shear structures.

Three-dimensional composite reinforcement through stitching creates a significant advantage for postbuckled shear panels.[4–6] Although stitching through the flanges of stiffeners will not improve their structural stability, it will help keep stiffness on the skin, thus taking higher loads on the skin. In addition, it will suppress the possible skin rupture precipitated by interlaminar failures. For these reasons the testing of two stack (0.090-in. or 0.229 cm) thick panels, with 10.0-in. (25.4 cm) spacing between stringers and 15.0-in. (38.1 cm) spacing between frames, is performed in this study.

Fixture Design

Picture frame shear fixtures are one of the most popular testing facilities used for the determination of the shear strength of stiffened panels. In the past, typical picture frame shear fixtures have been plagued by excessive stress concentrations at the corners of the specimen.[7] This is because of the pinching of the corners by the scissors action from the framing members of the fixtures, referred to as the "nutcracker effect." In the absence of cutouts or significant damages, the panel failure is typically initiated in the specimen corners at a lower shear stress than the panel could withstand without these stress concentrations. In this study a Farley–Baker-type fixture[8] has been used to test the stitched and unstitched panels because it reduces the magnitude of these stress concentrations primarily through the careful choice of the pin location.

The corner pin location of a shear test was recognized as being one of the most critical components of this type of test.[9] References 8 and 9 were focused on the study of thin-skinned postbuckled composite helicopter fuselage structures. As a result of the research, a Farley–Baker fixture was developed for the testing of fuselage panels. The corner stress concentration relief does permit for the obtaining of higher shear allowables than that of the typical picture frame fixture.

The four items identified as critical to reduce these stress concentrations, to get a purer state of shear, are as follows[8]:

1) Corner fixture pins are moved inward to the corners of the test panel and are not penetrating the plane of the test specimen, that is, eight pins total.

2) The location of the fixture pin is to the closest bolt on the doubler.

3) Loading tab (bonded steel doubler) stiffness is 30 times greater than the composite test panel.

4) A circular corner notch detail is to reduce stress concentrations further.

Specimen Design

Two 29.5-in. (74.93 cm)-square flat panels were designed and fabricated from the AS4/3501-6 material system. The first was made with stitching, and the second without. Both panels have longerons in the horizontal direction and frames in the vertical, to break the panel to 10- × 15-in. (25.4 × 38.1 cm) at the center. The frames have slots in the web to allow the longerons to pass through. The test specimen design is shown in Fig. 1. The frame geometry is shown in Fig. 2. The longeron geometry is shown in Fig. 3, and strain gauge locations are shown in Fig. 4.

Analysis

Closed-Form Analysis

A closed-form diagonal tension analysis was conducted using the method in Ref. 10. The analysis is based on the isotropic diagonal

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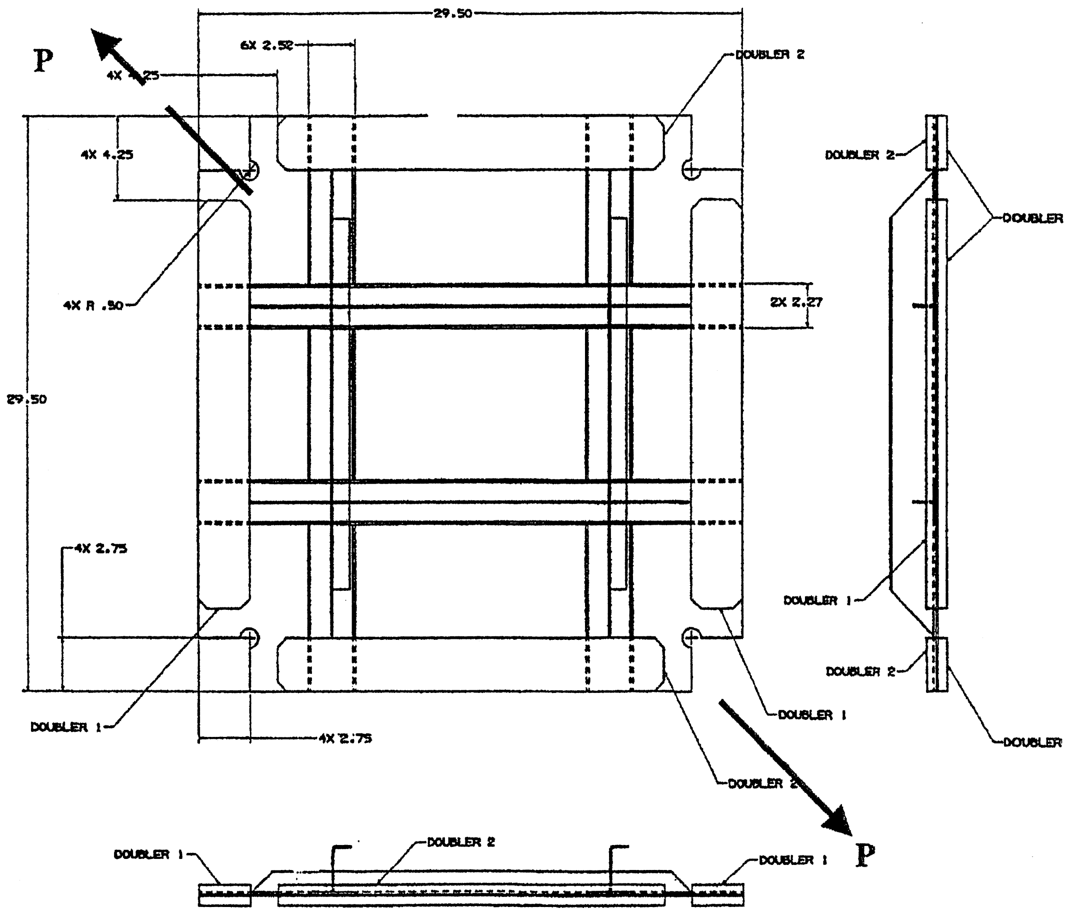

Fig. 1 Stiffened shear panel test specimen design.

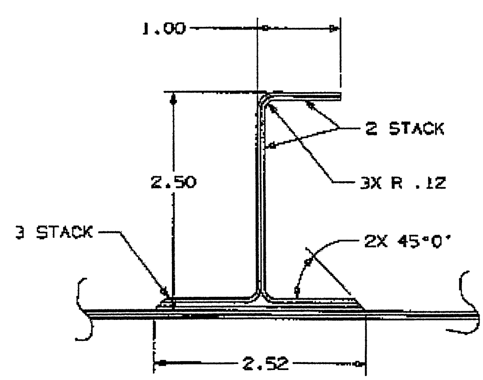

Fig. 2 Frame design.

tension theory.[11] The failure modes and secondary effects can be different for composite materials. The predictions in this study are therefore provided only as a comparison to the results of the finite element analysis.

The region of the shear test panel chosen for analysis was the center portion within the 10- × 15-in. (25.4 × 38.1 cm) section. The allowable local peak shear stress was calculated iteratively by using a spreadsheet. This local peak shear stress is

$$\tau = 38{,}426 \text{ psi } (265.04 \text{ MPa})$$

The local peak stress occurs at an applied shear running load of 3,226 lb/in. in the corner of the bay where the frame and stringer come together. As the frame and stringer deflected into the shear-deformed position, the pinched corners create a nutcracker effect. The global shear stress is 35,849 psi (247.26 MPa) at this time, which

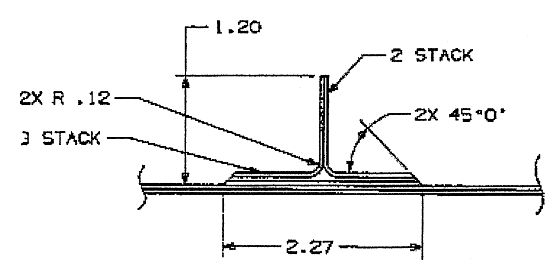

Fig. 3 Longeron design.

can be compared to an allowable stress of 43,594 psi (300.69 MPa) for this laminate. Morever, the predicted load for the onset of buckling, using the method from Ref. 7, is

$$N_{xy} = 344.7 \text{ lb/in. } (603.9 \text{ N/cm})$$

and this corresponds to an initial buckling stress of $\tau_b = 3{,}830$ psi (26.42 MPa).

Finite Element Analysis

A finite element model of the shear test specimen and fixture was created. This model was run using NASTRAN and ABAQUS codes. Linear and nonlinear buckling solutions were obtained. The NASTRAN buckling run predicted initial buckling to occur at an applied fixture tension load P_{cr} of 23,500 lb (104575 N). Final failure predictions were obtained from the nonlinear analysis. Nonlinear analysis shows a small inelastic behavior in load vs a displacement relationship. NASTRAN nonlinear analysis predicts the stiffened shear panel will fail at 71,700 lb (319065 N). ABAQUS nonlinear analysis shows almost identical findings. Table 1 lists the failure load predictions.

Maximum Principal Stress

Two-dimensional fringes from analysis results of NASTRAN and ABAQUS codes were done (top surface). NASTRAN nonlinear analysis shows a maximum principal stress of 56,300 psi (388.33 MPa), and ABAQUS nonlinear analysis shows a maximum principal stress of 56,590 psi (390.33 MPa). There are hardly any differences (0.5%) between these results. The tension allowable for this panel, made from laminated composite material, is approximately 69,300 psi (478 MPa) (average), and therefore the maximum principal stress is not a problem.

Minimum Principal Stress

Two-dimensional fringes from analysis results of NASTRAN and ABAQUS codes were completed (bottom surface). NASTRAN nonlinear analysis shows a minimum principal stress of −61,000 psi (420.74 MPa), and ABAQUS nonlinear analysis shows a minimum principal stress of −62,392 psi (430.34 MPa). There is a difference of only 2.3% between the results. The panel made from laminated composite material has an allowable tension of approximately −60,000 (average), and therefore the minimum principal stress is a problem. The ultimate failure will most likely occur in the +skin [10 × 15-in. (25.4 × 38.1-cm) panel] at the top or bottom corner.

Table 1 Finite element analysis failure load predictions

NASTRAN (linear analysis)	NASTRAN (nonlinear analysis)	ABAQUS (nonlinear analysis)
0.1-in. (0.25 cm) displacement 81,000 lb (360450 N)	0.1-in. (0.25 cm) displacement 71,700 lb (319065 N)	0.1-in. (0.25 cm) displacement 72,774 lb (323844 N)

Maximum Shear Stress

The maximum shear stress in the skin [10 × 15-in. (25.4 × 38.1 cm) panel] is at 37,600 psi (259.34 MPa) from the analysis results of the NASTRAN code, and the analysis results of the ABAQUS code were 1.2% higher at 38,066 psi (262.56 MPa). Overall, shear stresses from both results were analyzed, and strain gauges were located where the peak stresses were observed. The shear-stress fringe plots are somewhat different between results from NASTRAN and ABAQUS codes; however, the critical shear stresses are identical. The allowable shear of this laminated composite material

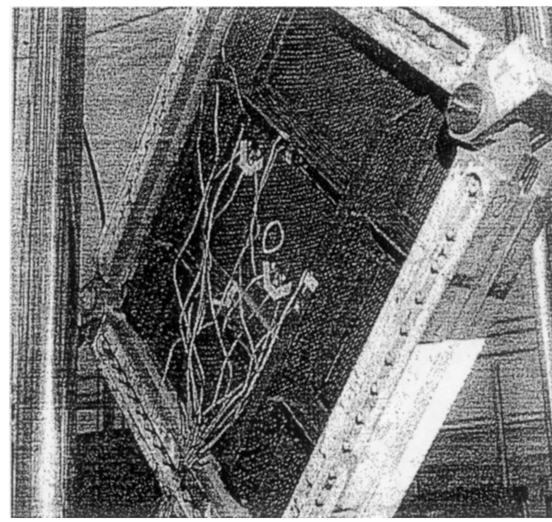

Fig. 5 Front of stitched panel after failure.

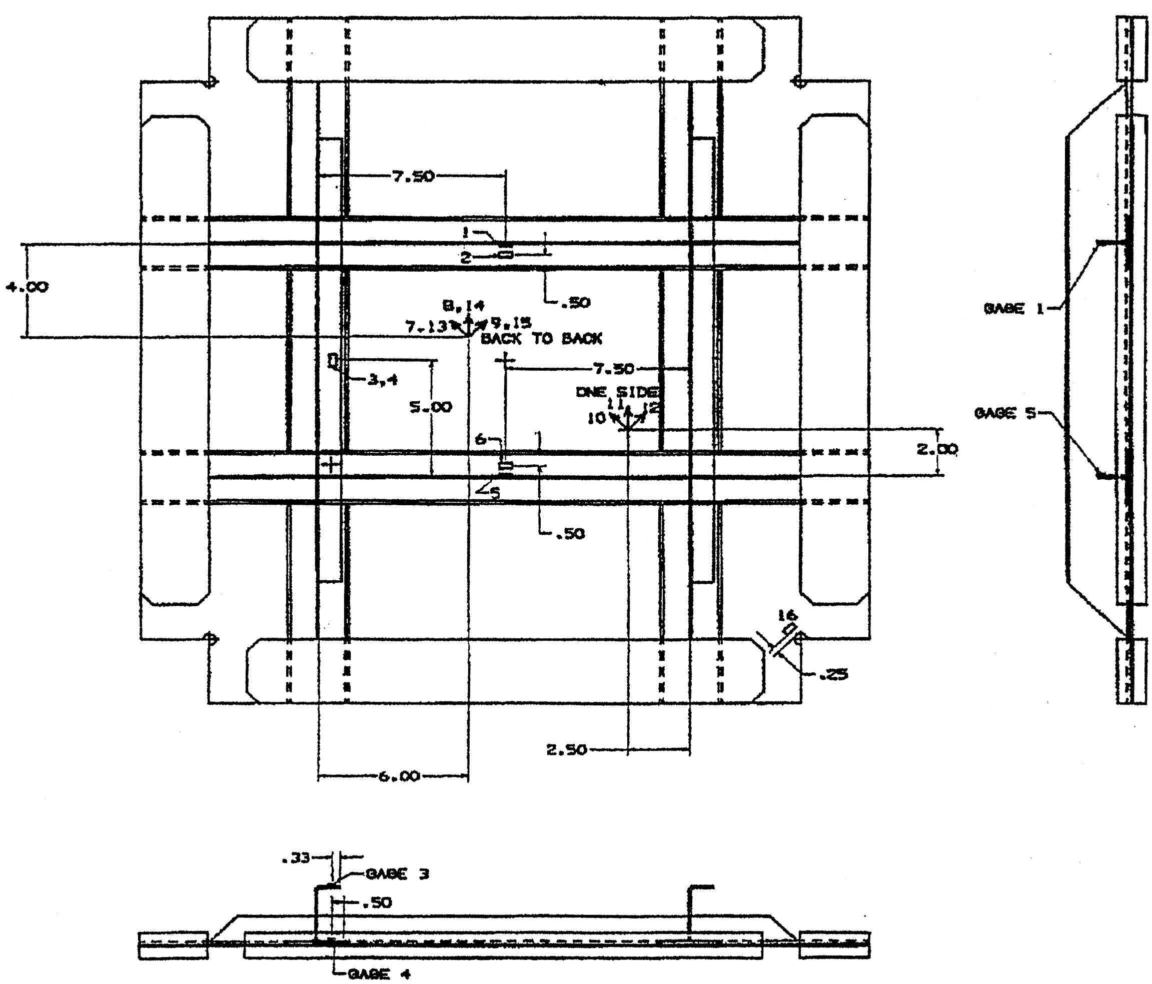

Fig. 4 Strain gauge locations.

Table 2 Shear flow in the panels

Specimen	Buckling shear flow, lb/in. (N/cm) Closed-form prediction	Finit element model prediction	Test results	Ultimate shear flow, lb/in. (N/cm) Closed-form prediction	Finit element model prediction	Test results
Stitched	345 (604.4)	692 (1212.4)	250 (438)	3226 (5652)	2113 (3702)	1805 (3162.4)
Unstitched	345 (604.4)	692 (1212.4)	405 (709.6)	——	——	1266 (2218)

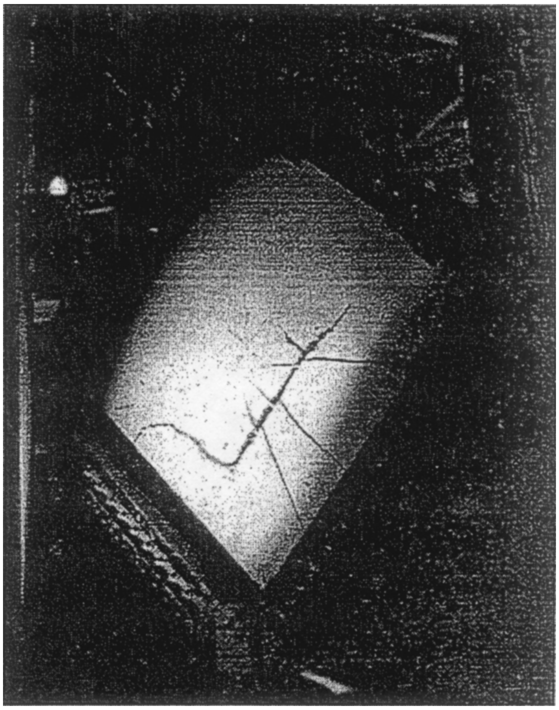

Fig. 6 Back of stitched panel after failure.

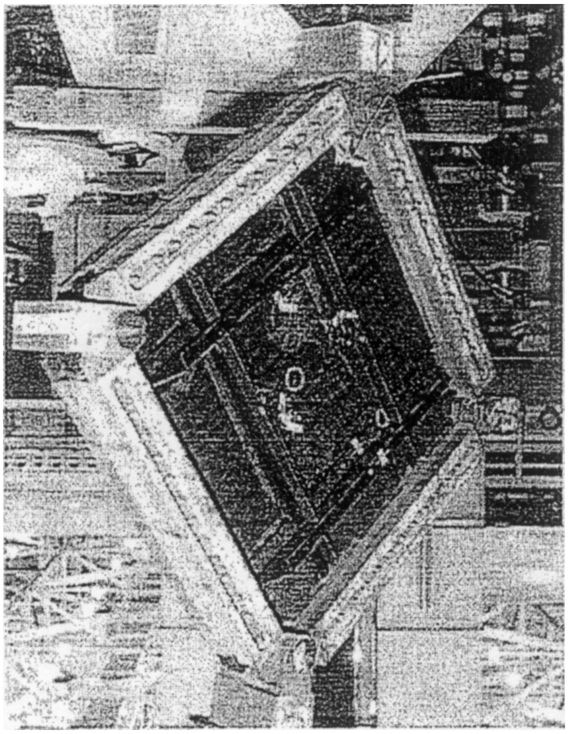

Fig. 7 Front of unstitched panel after failure.

panel is approximately 36,600 psi (252.44 MPa), and therefore the shear stress is a major concern indeed. The ultimate failure is most likely to occur in the skin at the top or bottom corner.

Maximum Principal Strain

The maximum principal strain of 70.00 μ-in./in. (μ-cm/cm) is the worst critical strain in the skin panel. The allowable strain in tension of this laminated composite material panel is approximately 95.00 μ-in./in. (μ-cm/cm) (average), and therefore the maximum principal strain is not a problem.

Minimum Principal Strain

The minimum principal strain of -89.70 μ-in./in. (μ-cm/cm) is the worst critical strain in the skin panel. The panel made from laminated composite material has allowable strain in the compression of -85.00 μ-in./in. (μ-cm/cm) and is therefore similar to minimum principal stress; the corresponding strain is a problem. The ultimate failure is most likely to occur in the skin at the top or the bottom corner.

Tests Results

Tests were conducted between unstitched and stitched panels until failure occurred. The data of the most concern to this study are the performance of the stitched panel. Therefore, attention will be focused on the stitched panel.

The stitched panel failed at a test fixture tension load of 61,249 lb (272558 N). It is converted to a running shear load by dividing the panel diagonal distance between pins.

$$N_{xy\,(\mathrm{ultimate})} = P/D = 61{,}249 \text{ lb}/\ 33.94 \text{ in.}$$
$$= 1{,}805 \text{ lb/in. } (3162.4 \text{ N/cm})$$

Table 2 is the summary of the test results and analytical predictions.

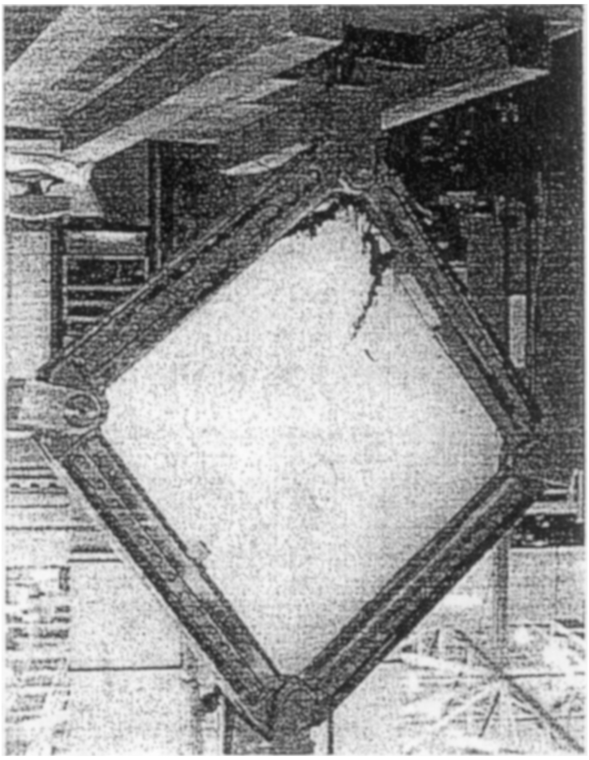

Fig. 8 Back of unstitched panel after failure.

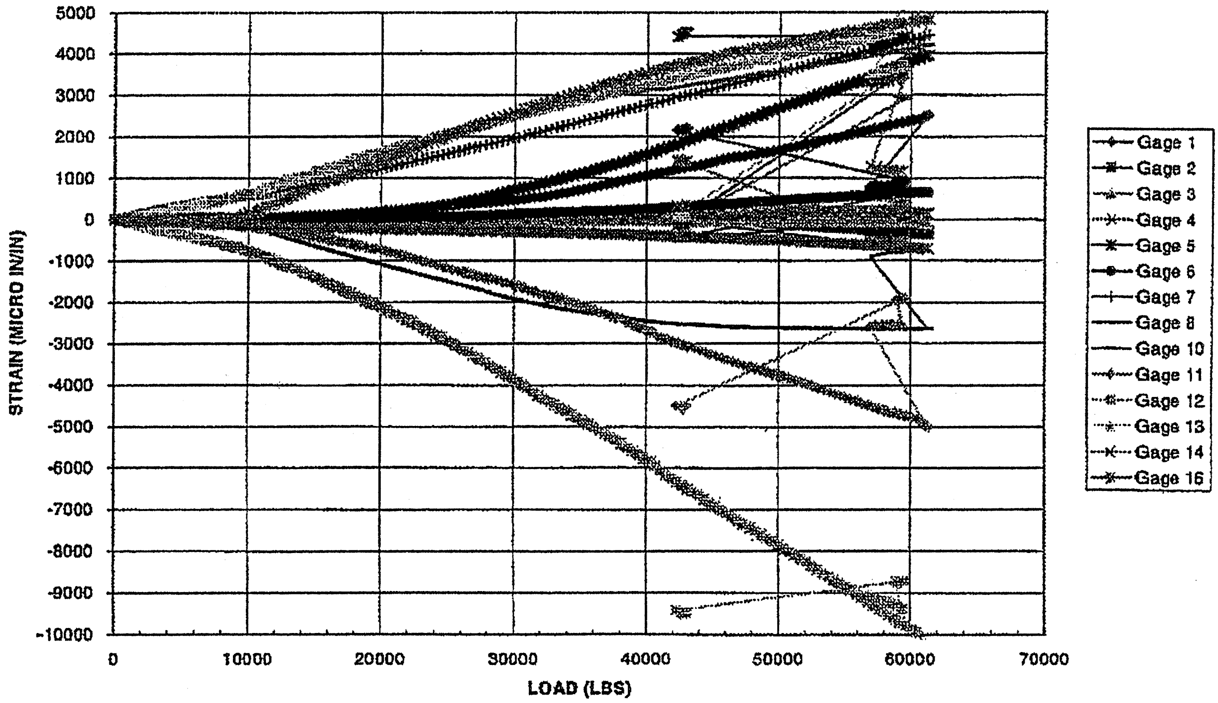

Fig. 9 Strain gauge data for the stitched panel.

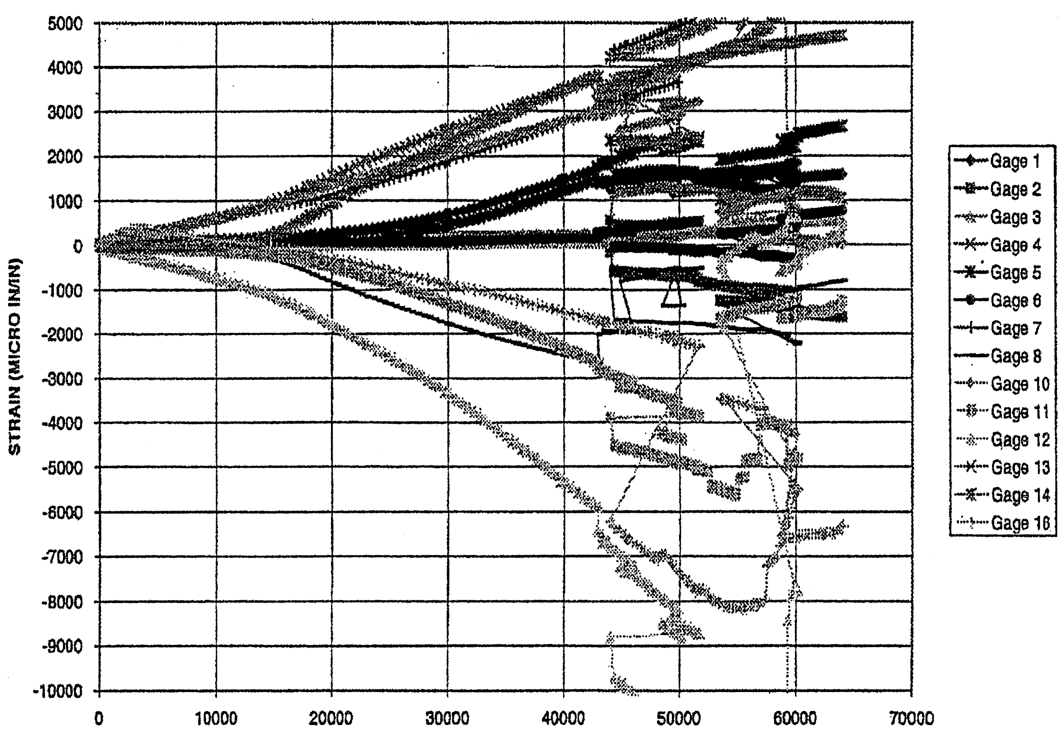

Fig. 10 Strain gauge data for the unstitched panel.

The load at which initial buckling occurs is seen to have significant scatter as the stitching process should not change this load. The predicted initial buckling loads, via the finite element analysis, are usually higher than that of the test results. This is because of the initial imperfections and eccentricities that can be introduced by the test fixture, as there is a small gap in the clevis. For the convenience of assembly, the test panel can be slightly mislocated from the centerline of the skin caused by this gap.

The closed-form solutions are expected to give lower initial buckling loads than the finite element solutions because of the definition of the size of the analyzed panel. In Ref. 11 the panel width is defined as the distance between longerons and frames and ignores the flange stiffness or additional restraint. However, the finite element solutions take these items into account.

If enough panels were tested, it is believed that the average initial buckling loads for stitched and unstitched panels would be virtually

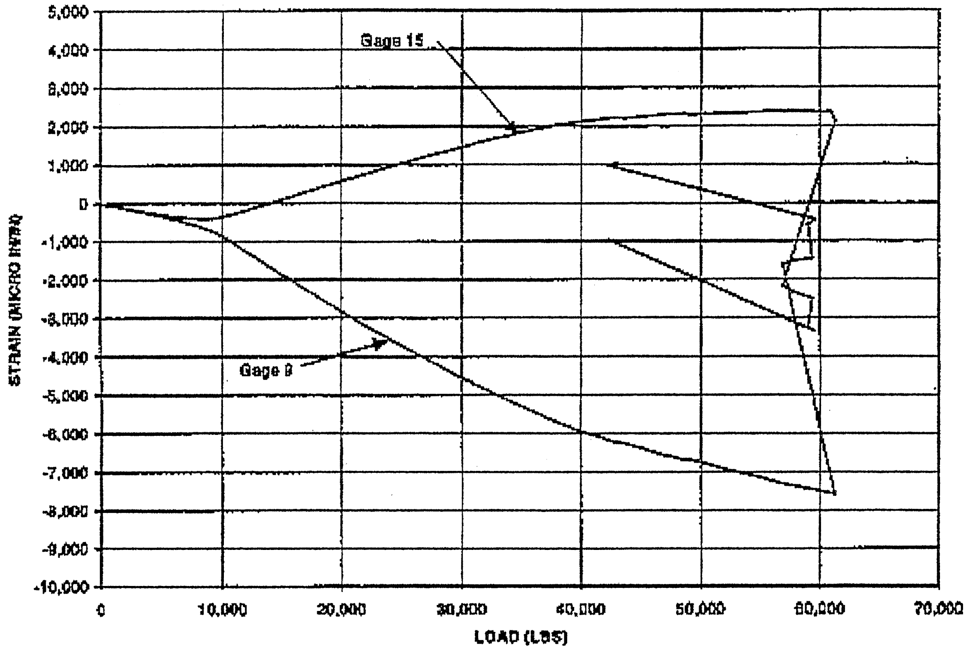

Fig. 11 Back-to-back strain gauge plot of stitched panel.

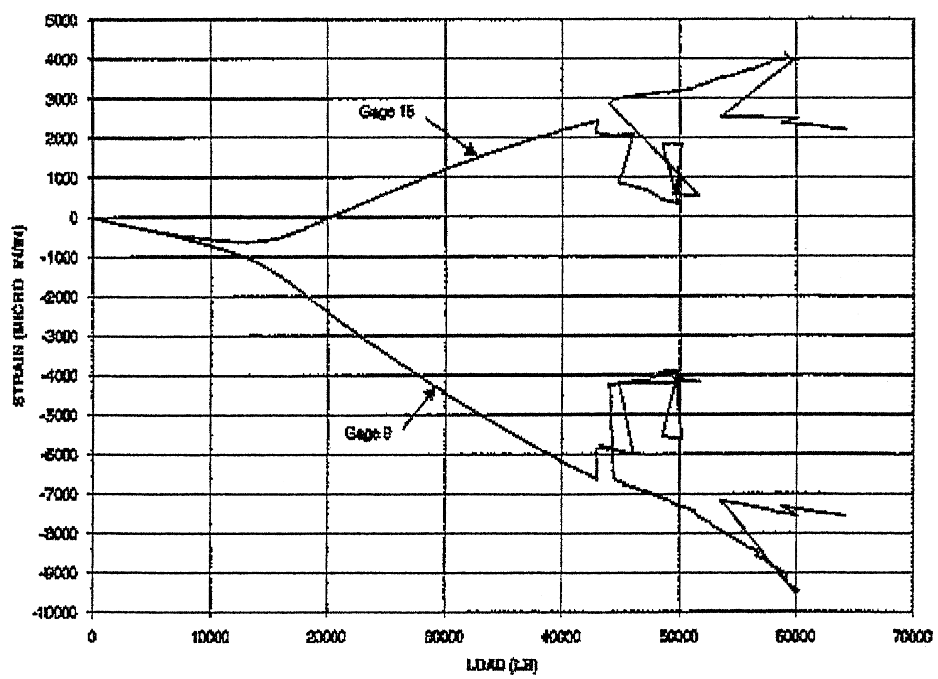

Fig. 12 Back-to-back strain gauge plot of unstitched panel.

identical. Increasing the doubler thickness to the maximum possible dimension in the fixture should minimize eccentricities and might reduce the scatter of the onset of initial buckling.

The stitched panel showed one of its primary strengths over the unstitched panel, which is the ability to stop the crack. At a load of 61,249 lb (272558 N), a crack was initiated and the load dropped. This crack stopped at the stiffener and did not propagate. The panel was then loaded to see how much more load-carrying capability was left. A load of nearly 60,000 lb (267000 N) was applied before resulting in a catastrophic failure. This failure appears to start at the crack, but then turned 90 deg along the stiffener until the frame was reached, where it turned another 90 deg to follow this frame flange. After nearly half of the skin section had been ripped off, the flange finally failed. The stitching, therefore, meets the design criteria by providing a crack stopper, which is not available in the traditional bonding of stiffeners. Stitching is also shown to dramatically improve the shear strength of the panel by preventing the premature debonding of the stiffeners.

Photographs of the failed test panels are shown in Figs. 5–8. Figures 5 and 6 are views of the stitched panel loaded in the testing machine. Figures 7 and 8 are views of the unstitched panel loaded in the testing machine. Figures 9 and 10 are strain gauge data of stitched and unstitched panels. Figures 11 and 12 are back-to-back strain gauge plots of stitched and unstitched panels.

Conclusions

This study is focused on the stitched and unstitched panels under shear loadings designed to reduce the weight of aircraft components. The skin on the AH-64 helicopter is loaded primarily in shear. To reduce structural weight, the aluminum skin is allowed to postbuckle in shear, a condition known commonly as diagonal tension. The postbuckled design is used to meet the aggressive weight goal.

This study shows that by using composite materials stitching the stiffener flanges to the panel skin will increase the shear strength capabilities of the panel up to 43%, as demonstrated by the test specimen. Moreover, a shear failure in the skin of the middle bay, within the stitched panel, shows that the skin-stiffener delamination failure mode was eliminated. It is well known that the skin-stiffener delamination failure is the primary failure model by the stitched/resin film-infused process.

It has been shown that there is significant scatter in the onset of buckling for this test configuration. Stitching is not believed to appreciably alter the initial buckling load of the test panel. Predictions based on the finite element analyses of this buckling load are usually higher than the actual bifurcation load because of initial imperfections and test fixture eccentricities.

Future shear panel tests should be fabricated with the maximum thickness doubler permitted by the fixture. This is to minimize the load eccentricities introduced into the panel by the slight mislocation of the test panel while assembling the fixture around it. Minimizing the load eccentricities in this manner can reduce the scatter of the critical load at which initial buckling occurs in different panels.

Acknowledgments

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References

[1]Acker, G. F., and Okamoto, F. S., "Design Criteria Report for the AH-64A Advanced Attack Helicopter Production," Hughes Helicopter, Rept. 77-Y-8001-2, July 1978.

[2]Kassapoglou, C., "Calculation of Stresses at Skin-Stiffener Interfaces of Composite Stiffened Panels Under Shear Loads," *International Journal of Solids and Structures*, Vol. 11, 1993, pp. 1491–1501.

[3]Kassapoglu, C., "Stress Determination at Skin/Stiffener Interfaces of Composite Stiffened Panels Under Generalized Loading," *Journal of Reinforced Plastics and Composites*, Vol. 13, 1994, pp. 555–572.

[4]Burwasser, J., and Hinrichs, S., "Composite Aircraft Fuselage Bead Stiffened Shear Panels," IRAD Final Rept. MDC93K0332, 20 April 1993.

[5]Rouse, M., "Effect of Cutouts or Low-Speed Impact Damage on the Postbuckling Behavior of Composite Plate Loaded in Shear," AIAA/Paper, April 1990.

[6]Taki, T., and Kitagawa, T., "Postbuckling Strength of Composite Stiffened Panel Under Shear Load," AIAA Paper, Sept. 1995.

[7]Bush, H. G., and Weller, T., "A Biaxial Method for In-Plane Shear Testing," NASA TM 74070, April 1978.

[8]Farley, G. L., and Baker, D. J., "In-Plane Shear Test of Thin Panels," *Experimental Mechanics*, Vol. 23, No. 1, 1983.

[9]Schneider, G. J., "Initial Buckling and Diagonal Tension Behavior of Kevlar/Graphite-Epoxy Stiffened Shear Panels," Sikorsky Engineering, Rept. SER-510047, April 1981.

[10]*The Boeing Company Information, Space and Defense Systems Strength Analysis Guide*, Boeing Co., 1985.

[11]Kuhn, P., Peterson, J., and Levin, L. R., "A Summary of Diagonal Tension Part I—Methods of Analysis," NACA TN 2661, May 1952.