Experimental Buckling of Cylindrical Composite Panels with Eccentrically Located Circular Delaminations

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An experimental investigation is carried out on the effects of circular delaminations (2, 3, and 4 in. in diameter) considering the buckling of Gr/Ep cylindrical panels. The results indicate a local instability within the delamination area followed by global panel buckling. The phenomenon of local instability (snap-through) varies with the delamination size.

Introduction

COMPOSITE materials are becoming increasingly popular for use in structures and applications where weight is critical. The wide use of composites has shown that there are problems not comparable to those of isotropic materials. One such problem and the subject of this paper is delaminations in the laminate. This problem has just recently received the attention needed to predict the behavior of the laminate. In addition, attempts have been made to describe the failure mechanisms of composite panel buckling in which delaminations are present.

One such attempt[1] used a one-dimensional analytical model to predict delamination buckling loads. Chai and Babcock[2] use a two-dimensional model to predict the loss of compressive strength due to a delamination brought about by a transverse impact. Shivakumar and Whitcomb[3] developed Rayleigh-Ritz and finite-element methods to determine the bifurcation buckling of an elliptical delamination embedded near the surface of a thick quasi-isotropic laminate under compressive loading.

Fei and Yin[4] analyzed a circular delamination in a thin, circular plate using the von Karman nonlinear plate equations. Their results indicate that a panel with a delamination under pure compression will only buckle, while a panel loaded in such a fashion that the delamination undergoes both compression and bending will experience buckling, snap-through (the separation of the delaminated layer from the main body of the plate), and catastrophic failure if the load is sufficient to attain the critical value for growth of the delamination. Experimental work has been done[5] with a delamination located in the center of a cylindrical composite panel which verified the usefulness of the Fei and Yin equations, even for a curved surface.

The purpose of the research reported herein was to experimentally determine the buckling loads of cylindrical composite panels with circular delaminations positioned near the surface of the panel. This is the same shape of delamination that results from an initial reaction to an impact load.[2] With the delamination placed near the surface of the cured panel, the snap-through effect[4] occurs. The experimental results are compared to finite-element analysis using STAGSC – 1[6] to help verify the experimental boundary conditions. One ply layup, one set of boundary conditions and one size panel, and three different delamination sizes were used in this study. The results produce a further understanding of composite buckling behavior and increase the data base available to users of composite materials.

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Panel Fabrication

The project[7] included the testing of 20 curved composite panels with a radius of 12 in. and a vertical length-chord width also of 12 in. containing a ply layup of $[0/\pm 45/90]_s$. Four panels of each of the following delamination types were manufactured: no delamination, 2 in., 3 in., and 4 in. The delamination was positioned between the second and third plies from the exterior (Fig. 1) of the side opposite the center of curvature. One additional set of four panels was fabricated with a 2-in. delamination at the midsurface of the laminate.

First, a proper delamination causing insert was needed. Teflon appeared to be the most effectively used material for this purpose. To further ensure a complete delamination, the teflon inserts were coated with RAM 225 release. The panels were laid up in molds (Fig. 2). There was one large panel that contained four test specimens per mold. A mylar template was used to place the teflon inserts in the correct locations, and the template was subsequently removed. The panels were then placed in the autoclave and cured in the molds. Upon removal from the autoclave, the large panels were C-scanned to ensure that the delaminations were in fact present and that they had not moved during the curing process. The test specimens were then cut from the large panels maintaining tight tolerances on the length dimension, since this proved to be critical[5] for the introduction of an even load distribution and to obtain a symmetric buckling pattern. Fifteen thickness measurements were taken and averaged for each panel.

All of the panels were manufactured from Hercules AS4/3501-6 graphite-epoxy 12-in. prepreg tape. The lamina material properties were experimentally determined to be

$$E_1 = 18.96 \times 10^6 \text{ psi} \quad G_{12} = .910 \times 10^6 \text{ psi}$$

$$E_2 = 1.48 \times 10^6 \text{ psi} \quad \nu_{12} = .280$$

Experimental Procedures

The compression test system consisted of a 30K-lb capacity machine with specially built supports to hold the panel (Fig. 3). The panel supports provided clamped top and bottom edges and simply supported sides. The top and bottom edges of the panels were clamped first, and the knife edges for the sides were applied last. The vertical sides of each panel had a strip of teflon tape applied to them to allow freer movement both vertically and circumferentially to help ensure the appropriate boundary conditions. In addition, "O" rings are included to assure a constant pressure against the test panel's vertical length (Fig. 4) and thus a uniform restraint.

The instrumentation consisted of four axial gages and a LVDT. The vertical LVDT was mounted in front on the crossheads to measure the end-shortening of the panel. Two strain gages were mounted 2 in. from the bottom and 4 in. from center, one on either side of the centerline, to check for uniform vertical loading. The remaining two gages were mounted back-to-back at the center of the panel and the delamination. Both the LVDT and the center strain gages were used to indicate the onset of buckling. The compressive load was introduced from the bottom.

Finite-Element Modeling

The panels were modeled using $2/3 \times 2/3$-in. square elements. The STAGS C-1 computer program does not have the capabilities to handle the three-dimensional characteristics of internal delaminations within a laminated panel, but it can be used for evaluating the bifurcation loads of a perfect panel. Thus for each panel experimentally in-

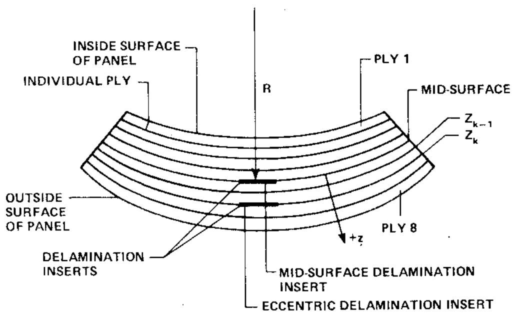

Fig. 1 Delamination location in ply layup of curved panels.

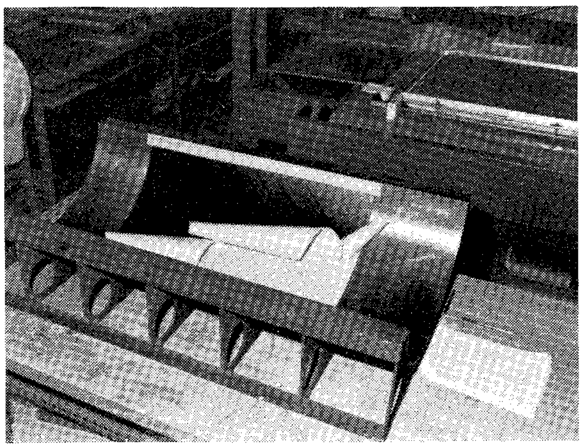

Fig. 2a Layup of small panels in one large mold.

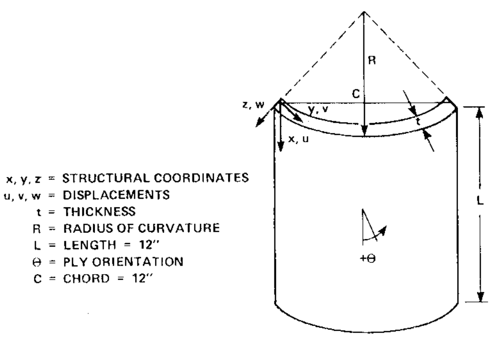

Fig. 2b Mylar template used to place delamination-causing insert.

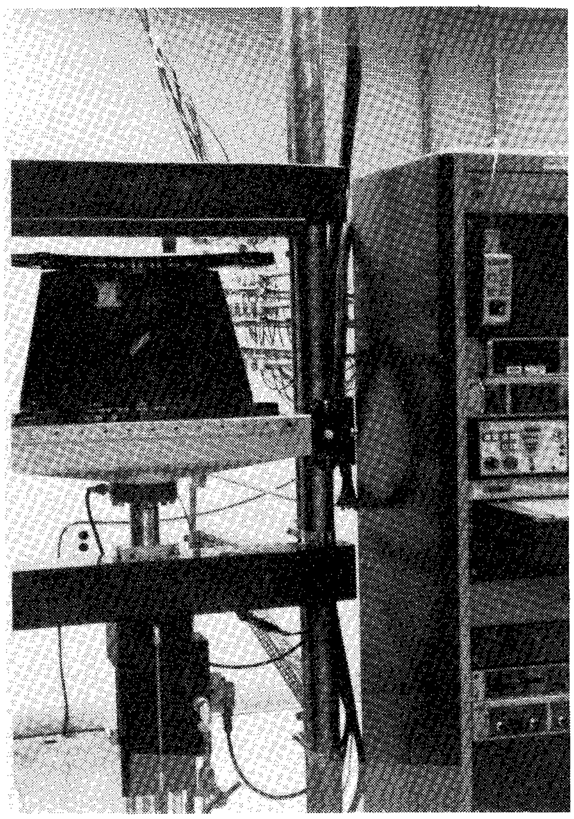

Fig. 3 Curved panel in test device.

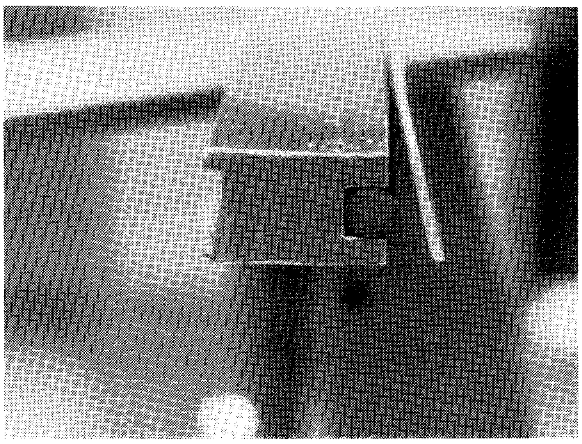

Fig. 4a Knife edge with "O" ring.

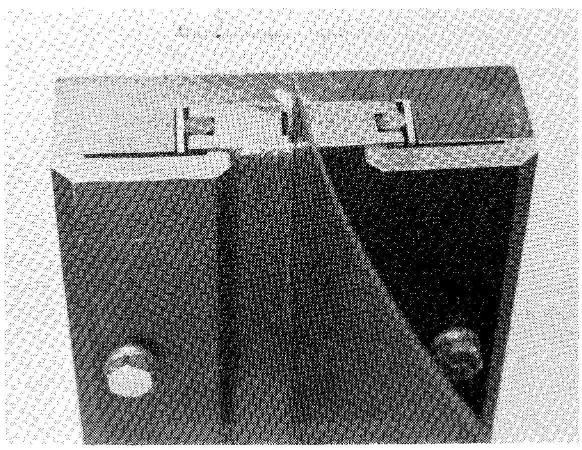

Fig. 4b Knife edge installed in vertical support.

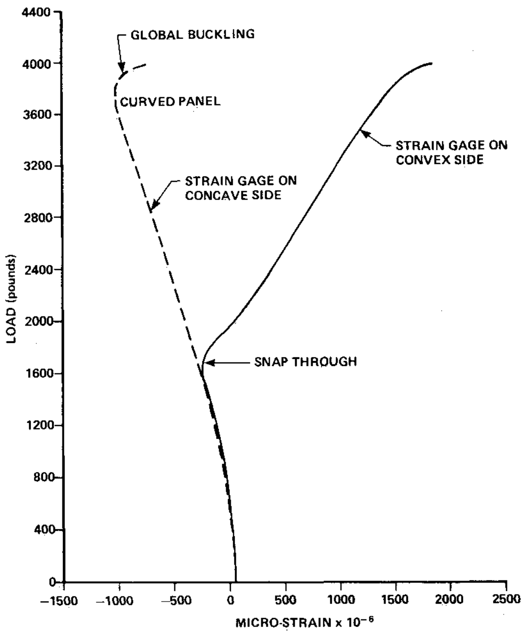

Fig. 5 Panel axial load vs strain reversed (2-in. delamination).

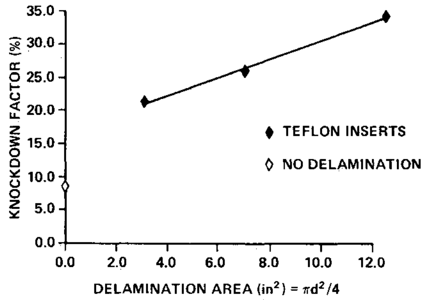

Fig. 6 Knockdown factor vs delamination area at global buckling.

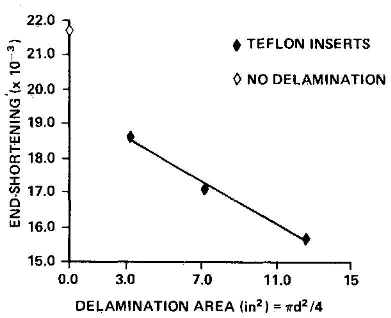

Fig. 7 End-shortening vs delaminated area at global buckling.

vestigated, a mean thickness was determined and a finite-element bifurcation load found. The test fixture did not completely allow free circumferential movement of the panel, and this effect was also investigated. The delamination experimental results were compared to the finite-element results using a function referred to as a knockdown factor where the

$$\text{knockdown factor} = \frac{100 \times (\text{finite element result} - \text{experimental result})}{\text{finite element result}}$$

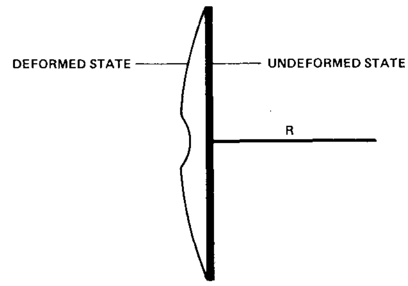

Fig. 8 Local buckling (2 in. delamination) vertical plane of symmetry.

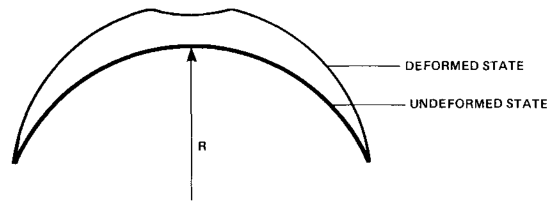

Fig. 9 Local buckling (2-in. delamination) horizontal plane of symmetry.

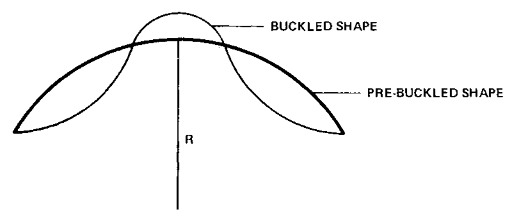

Fig. 10 Global panel buckling (horizontal plane of symmetry).

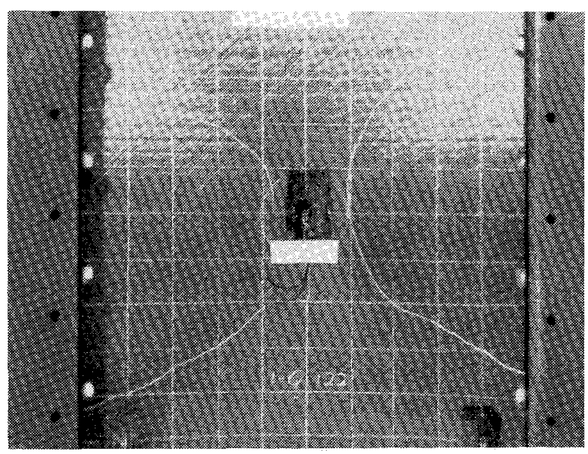

Fig. 11 Panel in its global buckling shape (2-in. delamination).

This factor, in effect, normalized the panels' imperfection and allowed for the specific delamination feature. The results from the finite-element solution established an upper bound to the overall experimentation.

Results

If one were to make a crude approximation based upon plate relationships with circular delaminations using the von Karman nonlinear expressions and strain energy release rates,[4] it is possible to roughly predict certain physical features of the delamination response. The physical snap-through or local buckling of the delaminated area occurred, and the delamination did not grow. These two phenomena could both be predicted by the work of Ref. 4, again within

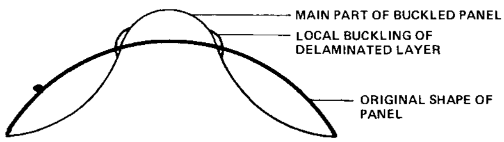

Fig. 12 Local buckling (4-in. delamination) horizontal plane of symmetry.

the accuracy of plate relations as applied to cylindrical panels. Figure 5 shows the actual snap-through for a 2-in. lamination in the cylindrical panel. The results of the panels with the delaminations in the center of the layup verified the earlier findings with no snap-through or delamination growth. The two experimental methods (the LVDT and center strain gages mentioned previously) for determining the onset of buckling yielded identical results.

The test fixture provided the boundary conditions for which it was designed. The panels with no delaminations present had a knockdown factor of only 8.3%. A knockdown factor this low for a buckling test indicates acceptable results, especially when considering all the variables that affect a compression test. From observing the testing, it is clear that the only boundary condition that may not have been correct is the free movement in the y direction. This was evident from the scarring on the teflon tape on the vertical sides in the y direction. The STAGSC-1 modeling with v fixed on the vertical sides showed 13% higher buckling loads than did the STAGSC-1 results with v free. The experimental results were lower than both of these results, which means that the knockdown factor was probably somewhere between 8.3% and 21.3%. Even with this 21% value, the results are favorable. For comparison purposes, it was decided to make comparisons with STAGSC-1 using the free circumferential boundary along the vertical edges.

The results of the panels with the teflon inserts revealed, as expected, an increasing knockdown factor with increasing delamination size. Figure 6 shows this effect with a plot of the knockdown factor vs the delamination area. The end-shortening of the panels with total delaminations also produces expected results. It can be observed in Fig. 7 that the panels with the larger delaminations do not deform axially before buckling as much as the panels with less delaminated area present do. This is a good indication that these particular panels have less strain energy present before buckling.

Buckling Patterns

To understand the buckling shape of the panels, an understanding of the panel movement prior to buckling is helpful. During loading, the panel is displaced away from the center of curvature. This movement is shown in Figs. 8 and 9. These figures show the movement of the vertical and horizontal planes of symmetry, respectively. The area where the delamination is located moved in the opposite direction toward the center of curvature, from the rest of the panel just prior to global buckling. This movement is detected with the center row of LVDT's. This movement is the snap-through effect that Fei and Yin predicted.[4]

The snap-through effect is not as obvious as it might have been with a flat plate. This is due to the circumferential stress developed from the deflection of the panel as the load is increased. The strain in the y direction was measured on each delaminated panel using a strain gage rosette in place of the normal axial gages in the center of the delamination. These measurements indicate the presence of a significant tensile strain on the outside surface or the side opposite the center of curvature of the delaminated layer and a smaller tensile strain on the inside surface. This large tensile strain restrained the delaminated area from snapping out away from the main body of the plate, which would require more strain energy. Instead, when the weakened delaminated area buckled locally prior to the global buckling of the panel, it moved toward the center of curvature or the path of least resistance and the smallest strain energy.

When the buckling load for the entire panel is reached, the panel buckles catastrophically. The sides move toward the center of curvature and the center of the panel moves away. Figure 10 shows the horizontal line of symmetry of the panel and the relative movement when the panel buckles. The actual buckled shape for a panel with a 2-in. delamination is shown in Fig. 11 and complies with the discussion related to Fig. 10. It is interesting to note that the 4-in. delaminated panels displayed a snap-through behavior in the buckled state. The 4-in. delamination extended beyond the center deformation away from the center of curvature (Fig. 12), and subsequently the edges of the delaminated layer were in a state of compression caused by the outer surface bending. This compressive force caused a small local buckle on the outer surface of the delamination. The buckling patterns for the panels with no delaminations and with total delaminations caused by the teflon inserts are very consistent with the shape of the panel in Fig. 12.

Conclusions

The following conclusions can be made from this experimentation:

1) Differences in the experimental panel thickness affect the buckling load.

2) Panels with delaminations located at the midsurface buckle at a higher load than panels with an off-center delamination.

3) The snap-through characteristics present with the larger delamination yield a complex local contact problem that until now has not been analytically considered.

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