

# POSTBUCKLING OF ECCENTRIC OPEN-SECTION STIFFENED COMPOSITE PANELS

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By

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## Abstract

A numerical study of the postbuckling behavior of open-section stiffened composite compression panels is presented with emphasis on the effects of an anisotropic attached flange on results, the strain distribution near collapse, and the change of buckle pattern during postbuckling response. Results are obtained for a blade stiffened panel with orthotropic or anisotropic attached flanges from a new version of STAGS, a general, branched shell, nonlinear computer program. Comparisons between these results indicate the effects of anisotropic flanges on the results. Elastic strain distributions are obtained from STAGS and indications are given about the collapse loads and modes of a particular panel. Change of buckle pattern is studied using STAGS and a special purpose computer program for long plates. Comparisons are made with experimental results.

## Introduction

Determination of the behavior of a compression-loaded stiffened structural panel constructed of metal or of laminated filamentary materials is an important design problem in aerospace applications. Stiffened panels having open-section stiffeners are generally easier to inspect and to fabricate than panels having closed-section stiffeners. Open-section stiffened panels are not as structurally efficient as closed-section stiffened panels, that is, they require more weight to carry a given load. Also, simple, closed form, column and local buckling formulas which treat various buckling modes independently are usually adequate for the design of closed-section configurations, such as the hat-stiffened configuration. More involved buckling analyses are required, however, to predict the buckling behavior of open-section configurations due to their more complicated stiffener twisting and rolling behavior. These various modes are interrelated and cannot be treated using simpler formulas for local buckling. Reasonably accurate buckling analysis has been derived and compared with other analyses and experiments for the buckling of open-section, laminated, stiffened panels (refs. 1 and 2).

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A number of other papers (e.g., ref. 3) have been written describing the postbuckling of stiffened panels based on results obtained from the general purpose computer program STAGS, which is described in reference 4. Also, since the publication of reference 3, additional capabilities have been added to STAGS (ref. 5). The new version is referred to as STAGS-TP. Three effects that have not been addressed in previous studies which are expected to be important are the effects of an anisotropic attached flange on results, the detailed strain distribution near collapse without considering an assumed initial imperfection, and the change in buckle pattern during the postbuckling response. A prebuckling, buckling and postbuckling study of open-section stiffened panels of laminated construction with emphasis on these three topics is presented in this paper.

### Anisotropic Attached Flange

Often to get desired layups in the stiffener and skin, there is no satisfactory alternative in manufacturing a stiffened panel but to make the attached flange anisotropic. A study is presented of the effects of using an anisotropic layup in the attached flange in conjunction with balanced symmetric orthotropic layups in the skin and blade stiffener of an eccentrically stiffened laminated panel. The study includes a comparison between the anisotropic flange panel postbuckling results and the corresponding balanced symmetric layup panel postbuckling results. The panel layups and cross-sectional dimensions are shown in figure 1. The panel length is 10 inches and the loaded edges are simply supported. The boundary conditions in the unloaded direction are chosen such that the panels are part of a continuous set that are connected side by side. Thus, two laminated panels made up of repeating segments are considered in the present paper, the cross sections of the repeating segments are indicated in figure 1. Both panels have the same layup in the skin and stiffener. Also, both have the same plies in the attached flange, but the arrangement is different. One has a balanced symmetric layup, the other does not which makes it anisotropic. Both have the same weight, the same thicknesses and, except for the details of the flange layup, the same eccentricities. They can be constructed and provide a reasonably good basis for a numerical study of the behavior of an anisotropic flange as compared to a balanced symmetric flange. Calculated results obtained from the STAGS-TP computer program for these two panels will now be discussed and compared.

The characteristic load-shortening curves (fig. 2) show that the buckling load for the panel with the anisotropic flange has a 10 percent smaller buckling load than the panel with a balanced symmetric flange. The initial postbuckling slopes are nearly equal, but the slopes are different for loads near twice the buckling load. At the higher loads the deflections are larger for the panel with the anisotropic flange as shown in figure 3. From a study of the computed data, the largest strains found were twisting strains near the loaded ends and bending strains in the unloaded direction at the midlength. Twisting strains near the loaded ends are shown in figure 4, and bending strains in the unloaded direction at the midlength are shown in

figure 5. The twisting strains in the panel as shown in figure 4 for the anisotropic flange is about 30 percent larger than the twisting strains for the panel with the balanced symmetric flange. The bending strains in the unloaded direction are not as large as the twisting strains. Large twisting strains in combination with such bending strains lead to delamination in blade stiffened panels which would indicate that the panel with the anisotropic flange would fail at a lower load than the corresponding panel with the balanced symmetric flange.

### Strain Distribution Near Collapse

With the present state-of-the-art of determining material properties and applying computer programs, it should be possible to determine in advance what to expect from a brittle test specimen. To test this hypothesis, a numerical study of buckling and of the strain distribution in the postbuckling range near collapse was needed. Test results for a panel that involved compressive buckling, postbuckling and collapse were obtained for panels from reference 1. The corresponding numerical strain distribution was obtained by analyzing the panel using STAGS-TP. New calculations obtained in the present study agree with buckling test results to within 1 percent. A photograph indicating the buckle pattern taken at the time of the experiment and a contour map of the deflections determined numerically are shown in figure 6. A comparison of longitudinal strain  $\epsilon_x$  at the strain gage in the bay as given by experiment and analysis is presented in figure 7 (unfortunately the strain gages are located on a node). Analysis is more conservative; there is good agreement in slope and experiment indicates that something happened at a strain of about .003--probably an early local failure or delamination. Only longitudinal strain was measured in the experiment. Analysis gives strain measures at many points throughout the model analyzed, and from these strains the maximum values were found to be all on the central stiffener. These results are plotted in figure 8, and they indicate high twisting and longitudinal bending strains near the extreme fibers in the postbuckling range. Delamination of the stringers should be expected, and that is what happened as indicated by the failed specimen as shown in the photograph, figure 9. As expected, meaningful comparison can be made for buckling, postbuckling and imminent collapse between present state-of-the-art elastic analysis and experiment with brittle materials

### Change in Buckle Pattern

The loading of a structure is along available equilibrium paths which tend to minimize the potential energy of the system. When the system is conservative there may be several paths available which give the same energy and the loading is then said to be path independent. For the postbuckling of a finite plate, the path of loading may involve smooth or explosive shifts or jumps in behavior as the plate changes from one buckle pattern to another. For an infinitely long plate the wavelength would change smoothly. Unless change in buckle pattern is accounted for, large errors in the analysis may

occur. To simplify the study of change in buckle pattern, an aluminum alloy plate supported along the length by knife edges instead of by stiffeners is considered and the apparatus, method of testing, and the results are presented in reference 6. Data were obtained for a 2024-T3 aluminum alloy panel 4.71 inches wide, 25.36 inches long (in the loaded direction) and .0708 inches thick. However, upon reexamination of the data, it was found that the characteristic curve presented in figure 3 of reference 6 is in error. Corrected curves are given in the present paper. The experimental results show that the plate buckled in five buckles, then upon increase in load, the mode changed to six buckles. The plate material started to behave nonlinearly (plastic behavior), then the mode changed to seven buckles with further increase of load, and then to eight buckles as shown in figure 10.

To study change in buckle pattern for aluminum or composite structures numerically, the STAGS-TP computer program (ref. 5) can be utilized. Clamped boundary conditions were considered for the loaded edges and zero inplane displacements were considered normal to the unloaded edges. A nonlinear computer solution indicated that inplane prebuckling deformations were present. A second computer solution using the method of reference 5 determined that the four lowest eigenvalues of the tangent stiffness matrix were very close together; the two lowest eigenvalues were within 2 percent of each other. Each of the paths of the lowest three modes was calculated up to about twice the buckling displacement as can be seen in figure 11. The path of the lowest mode has a jump in it at a load of about 590 pounds/inch. At this jump, the mode changes from two to four buckles. The path of the second lowest mode has a jump in it at a load of about 510 pounds/inch and then another at about 760 pounds/inch. The mode changes first from three to five and then from five to seven buckles. The path of the third lowest mode has a jump in it almost immediately after buckling where it changes from one to three buckles, then at about 560 pounds/inch it changes from 3 to 5 buckles. This form of analysis was then discontinued. It was recognized that the zero inplane displacement boundary conditions normal to the unloaded edges gave rise to stresses that were not present in the experiment. Thus for theory to have the same boundary conditions as experiment, the boundary conditions on the unloaded edges should have been less restrictive. A comparison of theory and experiment show that the slope of the lowest characteristic curve in the range where it represents 5 buckles is nearly the same slope as the experimental curve where it represents 5 buckles.

An alternate numerical approach was then attempted based on the consideration that an infinite plate analysis with unloaded edges free of stress would be suitable (ref. 7). For an infinite plate the changes in pattern would be smooth with continuous change in pattern permitted. Starting with 5 buckles at buckling, the number of buckles are identified on the numerical results curve in figure 11 by integers and the number of buckles is also identified on the experimental curve in the same figure. The experimental curve has a different initial slope than the numerical results because of imperfect matching of the edges and the testing machine platens. The slopes are nearly the same after buckling in the elastic range so that it

might be said that good agreement has been obtained between theory and experiment in the elastic postbuckling range for the load-shortening characteristic curve. Based on results in reference 6, it is expected that good results for the local strains and deformations are available from this infinite plate approach.

### Concluding Remarks

Three problems that occur for open-section stiffened composite or metal panels are studied in this paper. The first problem addresses the differences in the postbuckling results that are expected for a panel composed of balanced symmetric laminates and for the corresponding panel with an anisotropic attached flange. The second problem studies collapse as indicated from postbuckling elastic theory for a panel made from a brittle material as compared with experiment. The third problem studies change in buckle pattern as predicted numerically and comparison with experiment.

The panel with an anisotropic attached flange has a 10 percent lower buckling load and a lower stiffness at twice the buckling load than the corresponding orthotropic panel. Also, at twice the buckling load, the strains due to twisting are about 30 percent higher for the panel with the anisotropic flange. Numerical buckling results agree with experiment for the analysis of a brittle blade-stiffened laminated panel. At the failure load of the specimen, analysis gives strain measurements of the kind that agree with the experimental mode of failure, and the strains are large and increasing for load near failure. Numerical results are compared with experimental results for an aluminum panel which have changes in buckle pattern. Good agreement was obtained between theory and experiment in the elastic postbuckling range. Thus, this paper has demonstrated that the designer may obtain valuable information on the effects of anisotropic attached flange layup on mode of collapse and imminence of collapse of a stiffened panel, and may determine effects of change in buckle pattern for open section stiffened panels in the postbuckling range from available state-of-the-art computer codes.

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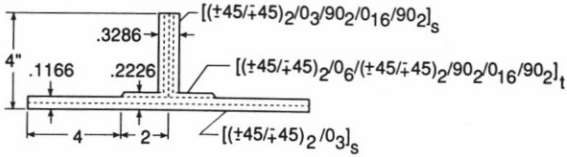
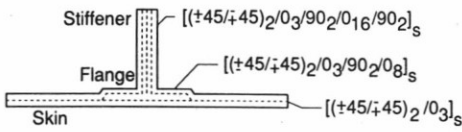


Figure 1. Cross-section dimensions and layups of eccentrically stiffened laminated panels.

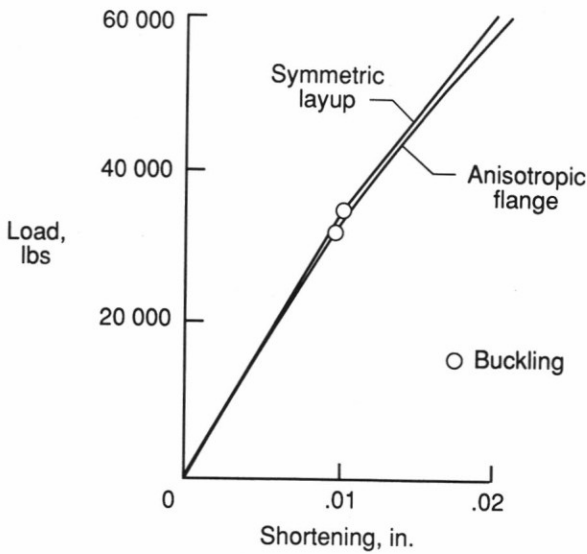


Figure 2. Characteristic load-shortening curves for eccentrically stiffened laminated panels.



Figure 3. Cross-sectional deflections at midlength of eccentrically stiffened laminated panels.

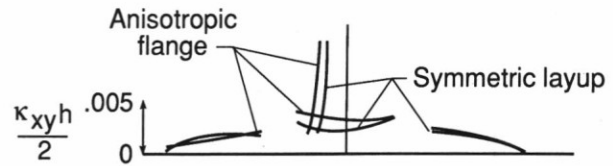


Figure 4. Distribution of twisting strains  $\kappa_{xy}h/2$  near loaded boundary of eccentrically stiffened laminated panel at a load of 60,000 pounds.

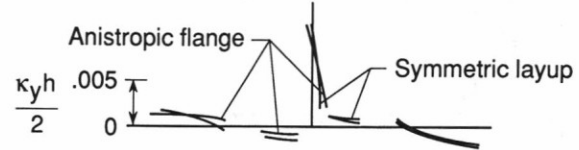


Figure 5. Cross-sectional distribution of bending strain  $\kappa_y h/2$  at midlength of eccentrically stiffened laminated panel at a load of 60,000 pounds.

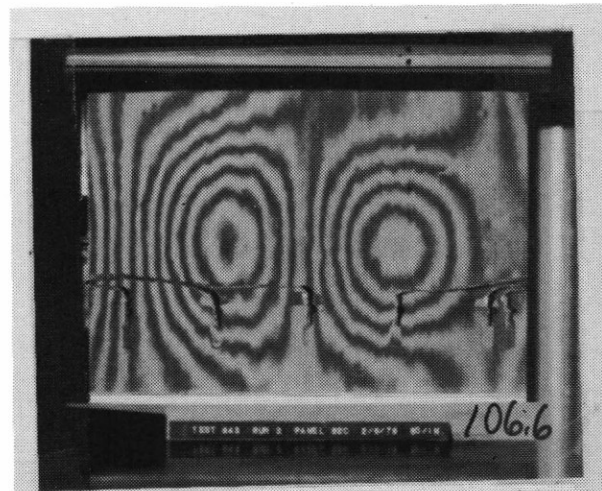


Figure 6. Moire fringes of the experiment indicating buckle pattern, and a contour map of the deflections determined numerically.

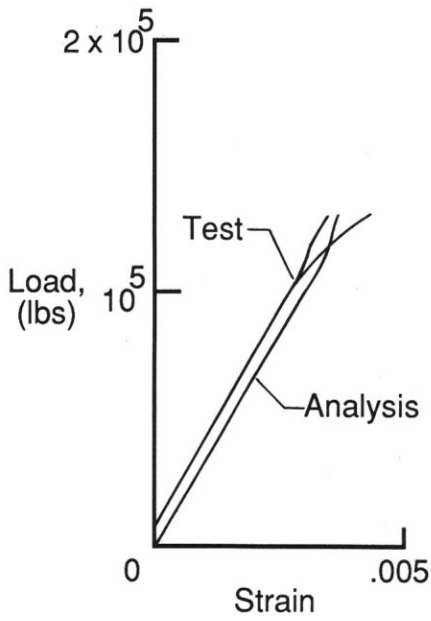


Figure 7. Strain in panel at the location of the gages as given by test and numerical analysis.

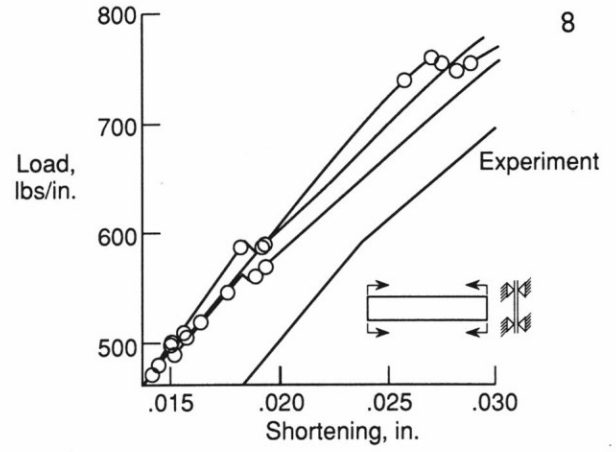


Figure 10. Calculated postbuckling curves for multi-bay panel with inplane deformation zero at the knife edge supports and experiment.

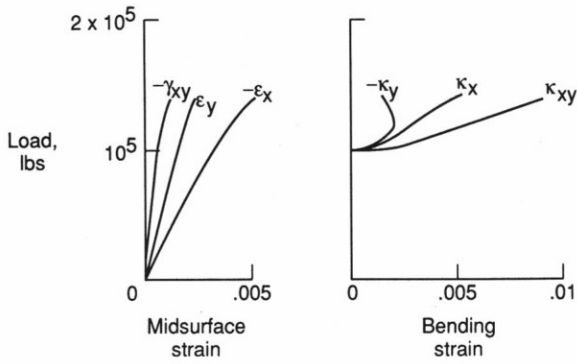


Figure 8. Maximum strains in panel as determined numerically.

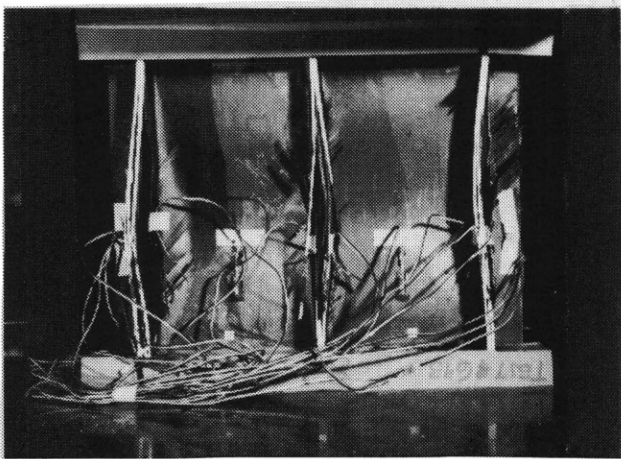


Figure 9. Photograph of panel after failure at a load of 130,000 pounds.

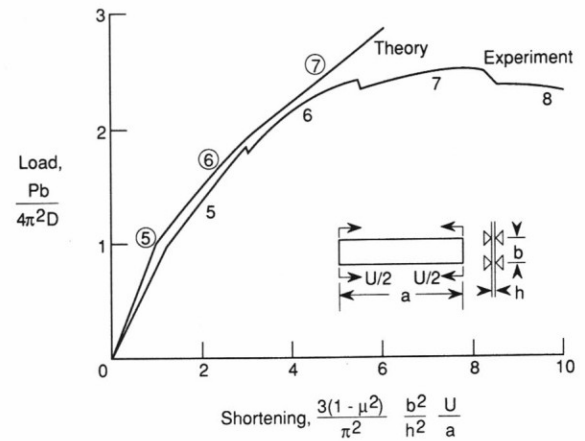


Figure 11. Comparisons of load-shortening curves a given by elastic theory and experiment.