NUMERICAL AND EXPERIMENTAL STUDY FOR FAILURE ANALYSIS OF LAYERED COMPOSITE STRUCTURES

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Numerical and experimental capability for prediction of nonlinear behavior and failure of layered composite panels subject to static loads is considered. This paper attemps to explain the mahanisms of failure of composite panel with experimental end theoretical evidence. The shell finite elements, based on higher order shear deformation theory, are combined with failure criteria to carry out initial failure analysis of layered composite structures. The computation results are quite coincident with the test results.

INTRODUCTION

The increased use of filamentary composite materials in airframe structures and the trend toward higher design strain levels have led to the need to understand the failure processes end to accurately predict the strength of structural composite laminates containing stress concentrators. In metals, it is common practice, particularly in lightly loaded regions, to allow such structures to develop postbuckling strength at proof or ultimate loads. This design philosophy has to be extended to composite structures if they are to remain competitive. The strain concentrations near holes cause reductions in the compressive strength of strength-critical laminates with holes. High interlaminar stresses at the edge of a uniformly loaded laminate have been known for same time (1) and shown to initiale failure.
Similar edge effects may be crucial to the postbuckling strength of the composite panel.
This paper attemps to explain the mechanism of failure of buckled, laminated plate, and supports the explanation with experimental and numerical evidence.
Nonlinear finite element analyses is used for the buckled laminated plate with holes. The purpose of the analysis for the prediction failure is in the evaluation of the stress resultants in the plate with a rectangle holes at the failure load.

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NUMERICAL ANALYSIS

The finite element method (FEM) has been used with much succes in predicting the response of both monolithic and composite material systems. Refined formulations (2) have been developed to characterize fundamental mechanical behavior of such composites, particularly with respect to quantifying transverse shear stress variations in laminated plate and shell type structures. Transverse effects are especialy significant for these materials systems because interlaminar strength are lower than the comparable in-plane values. The finite element approach in this work to represent buckling and postbuckling and behaviour of laminated composite structures, is based on formulation on utilising higher order shear deformation theory (3)-(4). The higher-order shear deformation theory used here takes into account the parabolic distribution of the transverse shear stress along the laminate thickness. The displacement field satisfies the condition that the transverse shear stress be zero on the plate surface and not zero in any other place. The displacement field is given by:

$$
\begin{aligned}
u_1 &= u + z\Big[\psi_y - \frac{4z^2}{3h^2}\Big(\frac{\partial w}{\partial x} + \psi_y\Big)\Big] \\
u_2 &= u + z\Big[-\psi_x - \frac{4z^2}{3h^2}\Big(\frac{\partial w}{\partial y} - \psi_x\Big)\Big] \qquad (1) \\
u_3 &= w
\end{aligned}
$$

where u, v and w are displacements of a point (x,y) of midplane and $\psi_x \psi_y$ are the rotations of the normals to the midplane about the axes x and y respectively.

FAILURE ANALYSIS

A finite element computational procedure is incorporated for the first-ply failure analysis of laminated composite shells. A number of existing failure criteria, including the maximum strain, Hill's, the Tsai-Wu and Hoffman's are included for linear or nonlinear analysis of laminated composite shells. Numerical-determined stress distribution in the postbuckling response were used with failure criteria to identify the load level and location of first-ply failure. In the present study, we asumed two failure modes could occur. These are matrix failure and fiber failure. In this case, the failed lamina is unloaded in transverse tension and shear, while continuing to carry loads in the fiber direction. If the fiber failure mode occurs, the lamina is assumed to be completely failed and the loads are transferred to the adjucent lamina. The occurence of the lamina failure mode was determined by using a failure mode criterion. Fiber failure was assumed to occur if

$$
\frac{K_1\Big[\int_{\varepsilon_1} \sigma_1 d\varepsilon_1\Big]}{\sum\limits_{i=1,2,6} K_i\Big[\int_{\varepsilon_i} \sigma_i d\varepsilon_i\Big]} \geq 0.1 \qquad (2)
$$

where $K_i = \left[\int_{\varepsilon_i} \sigma_i d\varepsilon_i\right]^{-1}$

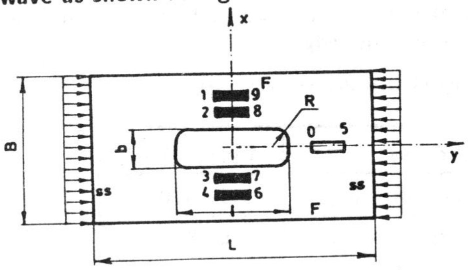
and failure criterion developed by Sandhu (5) was satisfied. Here ε_i and ε_{iu} are the current and ultimate normal (tensile or compressive as required) and shear strain components.

EXPERIMENTS

Compression tests to bucklings and then on to failure have been conducted on a range of plates with rectangle holes, Fig. 1. Three stacking sequences were chosen for the panels; 1 - $[0_2^\circ/\pm 45^\circ/0_2^\circ/0^\circ]_s$, 2 - $[\pm 45^\circ/0_2^\circ/0^\circ]_s$ and 3 - $[0_3^\circ/\pm 45^\circ/0_3^\circ/0^\circ]_s$. The panels were held in test machine with free sides and simply supported ends. Panels were fitted with four pairs of back-to-back resistance strain gages distributed across the specimen on a line passing through the hole center. Displacement transducers were used to measure both the axial contraction and out of plane deflection of the panel. A travelling displacement transducer could be traversed along the panel length to record the buckled mode shape. As well as measuring strains and displacements the acoustic emission from panels monitoring to provide a warning that a failure mechanism was active. This warning was crucial as it allowed the test to be stopped before compression-collapse destroyed the evidence. The damaged panels were then examed using ultrasonic techniques before subsequent re-loading and re-examination, until eventual collapse occured.

RESULTS

The results for behavior of panels with a rectangle holes subject to axial compression are shown in Figs 2 to 3. These panels buckle into one half-wave as shown in Fig. 2.



L=240 mm	E_{11}=145 kN/mm^2
l= 70 mm	E_{22}=10.70 kN/mm^2
B=120 mm	G_{12}=4.50 kN/mm^2
b= 25 mm	ν_{12}=0.31
R= 6 mm	t_1=0.125 mm

Figure 1 Description of axial compressed panel with rectangle hole

Ultrasonic examination of the panels at the onset of acoustic activity revealed very localised areas of delamination at the panel edges in positions corresponding to the buckle node lines, Fig 2.

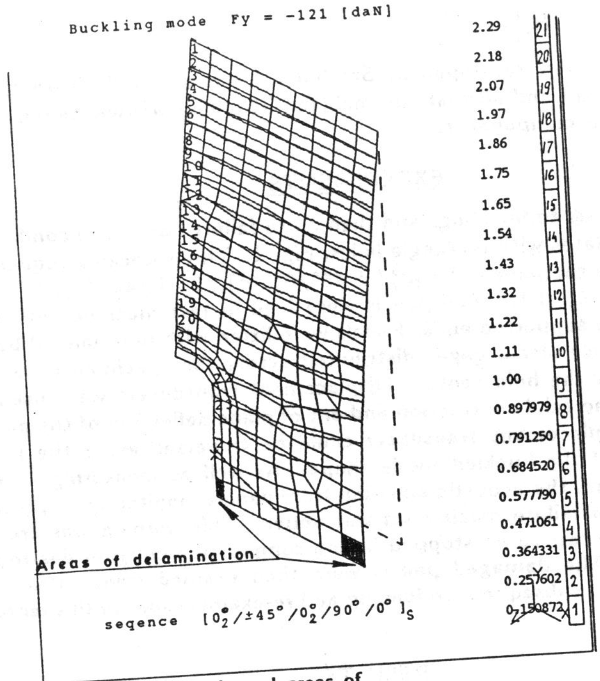


Figure 2 Buckling mode and areas of delamination for panel

These areas of delamination always occured between specific plies in the laminate, depending upon the stacking sequence. Fractographich analysis of a delaminated region indicated that failure was caused by an interlaminar shear stress τ_{yz} acting paralel to the panel edge.

TABLE 1: Buckling loads of composite plate with rectangle holes

	Sequences	F_{YCR} [kN]
1	$[0^\circ_2/\pm 45^\circ/0^\circ_2/90^\circ/0^\circ]_s$	1.2570
2	$[\pm 45^\circ/0^\circ_2/90^\circ/0^\circ]_s$	0.2815
3	$[0^\circ_3/\pm 45^\circ/0^\circ_3/90^\circ/0^\circ]_s$	2.6740

In Fig. 3 interlaminar shear strength $\tau_B = 80$ N/mm^2 is divided by calculated interlaminar shear stress τ_{yz}. A resonable agrement between experimental and numerical results is evident.

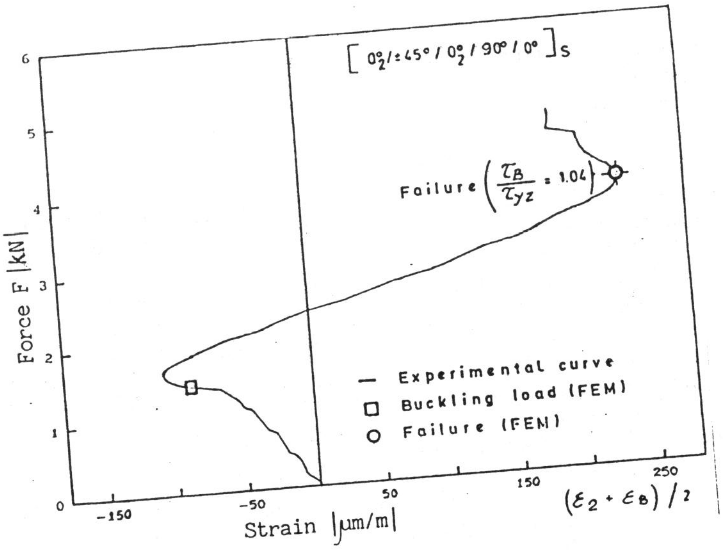


Figure 3 **Postbuckling behavior of panel with a rectangle hole - numerical and experimental results**

CONCLUSIONS

The analysis and mechanism for collapse of compression laminated panels with a rectangle holes are investigated experimentally and theoretically. The presented analysis found that the dominant stress in the boundary layer in all cases was interlaminar shear stress τ_{yz}. It also found that this stress had a maximum value at the ply interface where delamination was seen in the experiments, and that the change of sign of M_{xy} from one node line edge to the other accounted for the movement of the delamination across the mid-plane. The shell finite elements based on higher-order shear deformation theory give improved interlaminar shear distributions. A resonable agreement between nonlinear finite element solutions and experiments suggests a rational method for predicting failure loads.

SIMBOLS USED

E_{11}, E_{12} = longitudinal and transverse Young's modulus (kN/mm^2)
G_{12} = shear modulus (kN/mm^2)
F_{ycr} = buckling load of panel (kN)
t_l = ply thickness (mm)
τ_{xz}, τ_{yz} = interlaminar shear stresses (N/mm^2)
τ_B = interlaminar shear strength (N/mm^2)

ν_{12} = major Poisson's ratio
ε = strain
σ = stress

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