Effect of delamination on buckling strength of unidirectional glass-carbon hybrid laminates

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An experimental study is designed to understand the effect of delamination on critical buckling load and mode of failure for glass/carbon hybrid laminates. Firstly, delaminated specimens are compressed using a universal testing machine and buckling tests performed after the ends of each specimen are hinged to a special fixture designed to permit rotational boundary conditions. Non-linear buckling analysis based on incremental iteration method is conducted using finite element analysis (FEA) package (ANSYS) after conducting tensile tests to find the material properties. Critical buckling loads are found for symmetric \([C0/G45/G-45/G90]s\) and anti-symmetric \([C0/G45/G45/G90//G90/G45/G-45/C0]\) laminate configurations with four different delamination lengths \((a/L = 0.2, 0.3, 0.4 \text{ and } 0.5)\) created at two different delamination positions \((t/h = 0.5 \text{ and } 0.25)\). In this study, the critical buckling load decreased as the delamination length increased. Moreover, the critical buckling load tended to decrease when the delamination approached the surface.

Keywords: Buckling, Delamination, Glass/Carbon fiber, Non-linear analysis, Buckling mode

Fiber-reinforced composites in the form of relatively thin plates are used extensively. The load-carrying capability of composite plates against buckling under various loadings and boundary conditions have been well studied\(^1\). However, most of these studies have focused on single-fiber laminates, and hybrid fibers with their growing demands in applications such as windmill blades, aircraft structure and automobile parts deserve to be given higher research attention. Delamination is one of the most common failure modes in laminated composites, primarily resulting from manufacturing faults such as local lack of resin, impact of foreign object and void formation. Delamination can reduce the ability of a laminate to resist compressive load, leading to buckling at lower levels of compressive load. The critical buckling load also depends on area, shape and position of delamination.

Hwang and Mao\(^1,4\) investigated the effect of delamination growth on the compressive failure of laminated composite by non-linear buckling analysis. Using the total strain-energy release rate, the authors predicted the critical buckling load numerically and confirmed experimentally through compression tests. They found the buckling load to reduce when the carbon fiber layer was replaced with glass fiber layer with specific delamination length, buckling mode and stacking sequence. Nasr Esfahani \(et al.\)\(^5\) placed artificial delamination in three different positions and observed that the buckling load decreased as the delamination position approached the outer surface of the specimen.

Aslan and Sahin\(^6\) investigated delamination on the critical buckling load of E-glass/epoxy composite laminate containing multiple, large delaminations and found the longest and near-surface delamination size to influence buckling load and compressive failure load. In studying the effect of cutout, boundary condition and anti-symmetric configuration on buckling behavior of rectangular composite plates, Baba and Baltaci\(^2\) found that the buckling loads of anti-symmetrical laminates were higher as compared to symmetric ones and that the buckling load for clamped boundary conditions was higher than that for simply supported conditions. The micromechanical behaviour of hybrid fiber-reinforced composite lamina (graphite and boron fiber embedded in epoxy matrix) was studied by Sivajibabu \(et al.\)\(^7\) who reported that normal stresses at the fiber-matrix interface of the hybrid laminate, computed using FEA, were

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maximum at zero degree when the direction of load was normal to the surface.

In studying the effects of delamination width on the buckling loads of simply supported carbon/epoxy-woven laminated composite plates, Mehmet found that a decrease in the buckling loads occurred only when the delamination length-to-width ratio was above 0.1. With regard to the effect of aspect ratio on buckling of composite plates, Shrivastava and Singh found that at an aspect ratio <0.5 the plates failed by crushing and not by buckling. Kumar and Singh reported a FEA study of the effects of boundary conditions on buckling and postbuckling behavior of axially compressed quasi-isotropic laminate. They observed that the laminates that were clamped and those simply supported on all edges had the highest and lowest buckling and postbuckling strengths, respectively.

Hasan and Gokmen developed two-dimensional FE models using shell and contact elements in ANSYS to study the effect of delamination length and orientation angle on the natural frequency of symmetric composite beams. An innovative approach to create an analytical model using the Euler-Bernoulli beam and classical lamination theory (CLT) was proposed by Yap and Chai to obtain the buckling load of a delaminated composite beam. This analytical model was validated using FE method in which an 8-node shell element and a contact pair were used. Zhang and Wang presented a new mathematical model based on layer-wise plate theory to study compressive failure of laminate composite material having delamination.

A numerical study of linear and non-linear buckling analysis on unidirectional and cross-ply laminated plate using contact elements at the areas of delamination was performed by Cappello and Tumino. Although the influence of small delamination (a/L < 0.2) on critical load was negligible, delamination length, position and stacking sequence had higher impact on the buckling load and its mode shape.

The present study aimed to experimentally and numerically assess the effect of delamination length and position on the critical buckling load of hybrid composite laminate and its mode of failure.

**Materials and Methods**

Hybrid laminate composite plates were fabricated from unidirectional 400 gsm glass fiber and unidirectional 300 gsm carbon fiber. Epoxy resin LY556 was used as matrix material. The laminates were developed by hand lay-up technique in an open mold and cured in two stages – under atmospheric condition for 4 h followed by compressing in a compression molding machine for 10 min at 70°C under a constant pressure of 8 bars. Delamination of different lengths and positions were created artificially by inserting a Teflon film into the laminates during their making.

A symmetric laminate with orientation \([C_0/G_{45}/G_{-45}/G_{90}]_s\) and composed of two plies of carbon fiber and six plies of glass fiber was preferred in this study. To the glass fiber core, a high-modulus carbon fiber was placed as skin, with fibers distributed in all directions as against zero degree-oriented fibers employed in previous studies. Another (anti-symmetric) laminate with orientation \([C_0/G_{45}/G_{90}/G_{45}/G_{90}/G_{45}/C_0]_s\) was included to understand the effect of modifying 45° lamina to −45° lamina. The specimen with delamination is illustrated in Fig. 1, where \(a\) is the length of delamination and \(t\) is the depth of delamination. Gauge length \((L)\), width \((W)\) and thickness \((h)\) of the specimen were 200 mm, 40 mm and 3.6 mm, respectively. Specimens were fabricated with four different delamination lengths \((a/L = 0.2, 0.3, 0.4\) and 0.5) and two different delamination positions \((t/h = 0.5\) and 0.25).

**Testing**

The hybrid fiber-reinforced composite laminates were compressed using an Instron 3669 universal testing machine (UTM) with 50 kN capacity. The ends of the specimen were hinged using a special fixture as shown in Fig. 2. The fixture was so designed that the ends of the laminate were restricted to move only in the loading direction and rotate about the width of the specimen, thus ensuring hinged end condition. One end of the fixture was connected to the vice of the UTM and the other end to the laminate. The longer ends of the laminate are left free.
The laminate was axially compressed by moving the upper head of the UTM downward at a constant crosshead speed of 0.5 mm/min until buckling, whereas the lower head was stationary. Load and displacement were recorded and plotted to calculate the critical buckling load for each laminate. The buckling mode observed for each laminate was also captured. A specimen being tested in the UTM is seen in Fig. 3.

Material properties

Tensile test was conducted on separate specimens to know the Young’s modulus, Shear modulus and Poisson’s ratio as per ASTM standard D3039/D3039M and then to perform numerical analysis (FEA). For this, the specimens were prepared using the same fiber and resin by hand lay-up technique, as explained earlier. A specimen with strain gauge bonded being tested in the UTM is seen in Fig. 4.

Longitudinal tensile modulus, $E_{11}$, and major Poisson’s ratio, $\nu_{12}$, were determined from 0° unidirectional laminates and transverse modulus, $E_{22}$, and minor Poisson’s ratio, $\nu_{21}$, were determined from 90° unidirectional laminates using electrical resistance strain gauges. Shear modulus, $G_{12}$, was obtained from literature. The properties of glass/epoxy and carbon/epoxy laminates are given in Table 1.

### Numerical Study

Non-linear buckling analysis was carried out to predict the critical buckling load and buckling mode of hybrid fiber-reinforced laminates using ANSYS 12. Geometric non-linearity was included in this study for accurate prediction of critical buckling load.

Composite laminates were modeled using 8-node multi-layer shell elements (shell 281) having six degrees of freedom at each node. The beam contained two sub-laminates to represent delamination. The nodes were joined at the interface except at the delamination region where contact pair is applied to

<table>
<thead>
<tr>
<th>Composite laminate</th>
<th>$E_{11}$ (GN/m²)</th>
<th>$E_{22}$ (GN/m²)</th>
<th>$\nu_{12}$</th>
<th>$\nu_{21}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Carbon/epoxy</td>
<td>170</td>
<td>15</td>
<td>0.41</td>
<td>0.04</td>
</tr>
<tr>
<td>Glass/epoxy</td>
<td>77</td>
<td>39</td>
<td>0.24</td>
<td>0.11</td>
</tr>
</tbody>
</table>

Table 1—Properties of carbon/epoxy and glass/epoxy laminates
avoid overlapping of sub-laminates. This meant that the two sub-laminates were free to move away from each other in the delaminated region, but constrained to move as a single laminate in the other regions. Figure 5 shows a FE model with delamination length \((a/L) \ 0.5\) constrained in the non-delaminated region and the boundary conditions.

The FE model developed had eight layers with independent material property and fiber orientation for each layer. For a model with delamination position \((t/h) \ 0.5\) (mid-plane delamination), both the sub-laminates would contain four layers on either side of the delamination. Whereas, for a model with delamination position 0.25 (near-surface delamination), one sub-laminate would contain two layers and the other would contain six layers on either side of the delamination (Fig. 6). For all the models, carbon/epoxy material property was defined for the outer layers and glass/epoxy material property was defined for the remaining layers.

To simulate the hinged end condition on the left and right edges, the displacements (x, y, z) and rotations \((\theta_x, \theta_z)\) of all nodes at the left edge and the displacements (y, z) and rotations \((\theta_x, \theta_z)\) of all nodes at the right edge were set to zero. Load was applied at the right edge in the x-direction. In non-linear buckling analysis, a small out-of-plane perturbation should be applied to initiate buckling, either as small out-of-plane displacement or out-of-plane load at a point where maximum deflection occurs. In our study, a small out-of-plane load was applied in the middle of laminate to initiate buckling.

Results and Discussion

Buckling of laminate with mid-plane delamination

Buckling loads were obtained by compressing the symmetric laminate \([C_0/G_{45}/G_{90}]/\), and anti-symmetric laminate \([C_0/G_{45}/G_{90}/G_{90}/G_{45}/G_{45}/C_0]_s\) at delamination lengths equal to 0.2, 0.3, 0.4 and 0.5 at the mid-plane \((t/h = 0.5)\). The load-displacement curve for a symmetric laminate with delamination length 0.4 is shown as an example in Fig. 7. Similar curves were obtained for all the laminates. The average critical buckling load was found for each delamination length for symmetric and anti-symmetric laminates.

The experimental critical buckling loads for symmetric and anti-symmetric laminates with mid-plane delamination for different delamination lengths is shown in Fig. 8. It was noted that the critical buckling load reduced with increasing delamination length, in agreement with other studies. In addition, the critical buckling loads for both symmetric and anti-symmetric laminates were nearly matching for all the cases. Hence the use of an anti-symmetric laminate presented no change in the critical buckling load. For the laminate with \(a/L = 0.2\), the critical buckling load was 2.45 kN, which reduced to 2.01 kN when \(a/L\) was increased to 0.5. This means that there was an 18% reduction in the critical buckling load when \(a/L\) was increased from 0.2 to 0.5. When the delamination length was increased from 0.2 to 0.3, the buckling...
load reduced by 3%, whereas it reduced by 9% when delamination length was increased from 0.3 to 0.4.

Critical buckling loads of symmetric and anti-symmetric laminates with mid-plane delamination were obtained numerically using FEA and found to be matching for all the cases. The numerically predicted critical buckling loads (by non-linear buckling analysis) were then validated experimentally. Fig. 9 compares critical buckling loads obtained experimentally and numerically (non-linear FEA) for the laminates containing mid-plane delamination of various lengths. The comparison shows good agreement between experimental and non-linear FEA results with a deviation of 3.14, 0.59, 1.16 and 2.39% for a/L ratios 0.2, 0.3, 0.4 and 0.5, respectively.

**Buckling of laminate with near-surface delamination**

Experimental buckling loads for the symmetric and anti-symmetric laminates with near-surface delamination ($t/h = 0.25$) and delamination length equal to 0.2, 0.3, 0.4 and 0.5 were obtained from compression tests (Fig. 11). The load-displacement curve for symmetric laminate with delamination length 0.2 is shown as an example in Fig. 10.

The experimental critical buckling loads for symmetric and anti-symmetric laminates with near-surface delamination for different delamination lengths are shown in Fig. 11. We note from this that the critical buckling load reduced with increasing delamination length and showed very small deviation between symmetric and anti-symmetric laminates, as was observed for laminates with mid-plane delamination. But critical buckling load reduced by 29% when delamination length was increased from 0.2 to 0.5, as compared to an 18% reduction in the case of mid-plane delamination. The critical buckling load was 2.105 kN for the laminate with $a/L$ equal to 0.2, and 1.49 kN for the laminate with $a/L$ equal to 0.5. When delamination length was increased from 0.2 to 0.3, the buckling load reduced by 9%, whereas it reduced by 14% when the delamination length was increased from 0.3 to 0.4. From these results, we could conclude that for a unit increase in delamination length, the percentage reduction in
buckling load was more as the delamination position approached the surface.

The critical buckling loads predicted numerically (non-linear buckling analysis) for laminates with near-surface delamination and different delamination lengths were validated experimentally. Figure 12 compares the critical buckling loads obtained experimentally and numerically (non-linear FEA) for the laminates containing near-surface delamination of various lengths. The comparison shows good agreement between experimental and non-linear FEA results with a deviation of 3.09, 1.82, 4.65 and 12.75% for \( a/L \) ratios 0.2, 0.3, 0.4 and 0.5, respectively.

**Buckling mode of laminates**

Three different buckling modes, namely global, mixed and local, were obtained experimentally and numerically. Global buckling mode occurred for laminates with small delamination length \((a/L \leq 0.3)\) irrespective of delamination position (Fig. 13). In this mode, both the sub-laminates buckled in same direction with same out-of-plane displacement. Mixed mode occurred for the laminates containing mid-plane delamination \((t/h = 0.5)\) and large delamination length \((a/L \geq 0.4)\), as shown in Fig. 14. Whereas, the local buckling mode occurred for laminates containing near-surface delamination \((t/h = 0.25)\) and large delamination length \((a/L \geq 0.4)\), as shown in Fig. 15. In the case of local buckling mode, we observed that the thinner sub-laminate buckled away from the other sub-laminate in the opposite direction.

**Effect of delamination position on critical buckling load**

From Fig. 16, we observe that the critical buckling load of laminates containing near-surface delamination is lesser than that of laminates with mid-plane delamination. For laminates with near-surface delamination and delamination length...
equal to 0.2, 0.3, 0.4 and 0.5, the critical buckling loads reduced by 14%, 19%, 23% and 26%, respectively. This suggests that the effect of delamination position may be more as the delamination length increases. From these results, we could conclude that the delamination position has a significant influence on the buckling of delaminated plates.

Conclusions

This study investigated the buckling behavior of glass/carbon hybrid fiber composite plates containing through-the-width delamination under in-plane compressive load. The influence of delamination length and its position on the critical buckling load and its mode of failure was the main focus.

From the experimental and FEA results, the following conclusions can be derived:

(i) Magnitude of critical buckling load could decrease by increasing the delamination length.
(ii) Buckling load of laminates with mid-plane delamination can be higher than that of laminates with near-surface delamination.
(iii) The effect of delamination position can be more for laminates with larger delamination length.
(iv) There could be no difference in the critical buckling loads between symmetric and anti-symmetric laminates.
(v) The critical buckling load reduced by 26% for laminate with near-surface delamination when the delamination length was equal to 0.5.
(vi) The critical buckling load reduced by 18% when the delamination length was increased from 0.2 to 0.5 for the laminate with mid-plane delamination, whereas it reduced by 29% for the laminate with near-surface delamination.
(vii) The experimental and non-linear FEA results were found to be in good agreement.
(viii) Increasing the delamination length would not only reduce the relative buckling load but also affect the buckling modes. Three different modes of buckling were observed, viz., global, mixed and local.
(ix) Global mode occurred when delamination length was equal to 0.2 and 0.3 for mid-plane and near-surface delamination, respectively. Mixed mode occurred in case of laminate with mid-plane delamination ($t/h = 0.5$) and large delamination lengths of 0.4 and 0.5, whereas local mode occurred in case of laminate with near-surface delamination ($t/h = 0.25$) and large delamination lengths of 0.4 and 0.5.

References