EFFECT OF MANUFACTURING DEFECTS AND THEIR UNCERTAINTIES ON STRENGTH AND STABILITY OF STIFFENED PANELS

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1 Abstract

The use of composites in aircraft structures requires an accurate assessment of effects of known manufacturing defects such as gaps, fiber waviness, and disbonds. Relying on test alone to evaluate such effects is costly, time consuming, and can delay entry of aircraft into market. Test validated advanced computational multi-scale and multi-physics damage tolerance approach is proposed to evaluate the effects of manufacturing defects. The approach is effective at various levels of the building block of structures for a fixed wing aircraft. The analytical approach was first validated with test data for laminate with defects. Then it was applied to the evaluation of notched laminates and stiffened panels. The results for notched laminates, like in the test, indicated that initially the gap reduced the strength of notched laminates but its effect diminished as the gap size grew. Gaps in stiffened panels were found to have minimal effect on stability and strength. Beside gap defects, composite flat panels with delaminations in form of disbonds were modeled and analyzed to evaluate their effects on stability response. Structural analysis and test of the flat panels indicated that initial delamination can adversely affect the stability of composite panels once the disbonds reached a critical size. Details are discussed next.

2 Introduction

Polymer composites have found many applications as advanced engineering materials due to their lightweight, fatigue and corrosion performance. Improvement of automated manufacturing processes made them more attractive. These materials are now essentially employed in primary aircraft structures. This trend is intensifying with automated fiber placement process allowing double contour part manufacturing with effective unidirectional material. With all these advancements, still certain types of defects in polymer composites are inadvertently produced during manufacturing and during deployment of composite parts into service. Typical defects encountered in manufacturing of composites are intraply overlaps, gap defects, void of different shape and size, and fiber waviness. Fig. 1 shows an example of gap pattern for automated fiber placement process [1]. Fig. 2 shows an example of composite gap defect in 90 deg ply.

Relying on test solely to evaluate such effects is costly, time consuming, and can delay aircraft certification and entry into market. To mitigate risk associated with potential composite manufacturing or in-service defects, the authors evaluated the benefits resulting from the application of advanced multi-scale/ multi-physics durability and damage tolerance approach to reduce testing and accelerate certification of aircraft structural components. The analytical approach combined micromechanics with finite element analysis, damage tracking and fracture, and lamina property prediction with and without manufacturing defects.

Sawicki and Minguet [2] evaluated the effects of overlap/ gap presence on notched and un-notched laminate compression strength using specimens containing defects of defined size and locations. They reported strength reductions up to 27% in laminates containing gaps and overlaps at least 0.03” (0.762 mm) wide. Their study determined that further reductions in strength were not observed.
when wider defects were present. The compression strength reduction for un-notched and notched laminates was the same and was mainly caused by out-of-plane waviness induced by the defects on the subsequent plies. Similarly, Hsiao and Daniel [3] investigated through analysis and experiments the effects of fiber waviness on compressive strength of composite laminates. Their work also concluded that in unidirectional composites the compression strength is degraded seriously with increasing fiber waviness due to interlaminar shear failure.

Other defects in the centre of composite materials research are initial delaminations that may take place during manufacturing or as a result of object impacts, or high stress concentration. Wang et al. [4] investigated the effects of delamination size and delamination depth on stability of composite panels. They concluded that the extremely large delamination size resulted in a decrease in the plate resistance against the buckling behavior.

The work presented in this paper evaluated effects from multiple defect characteristics including delamination size, location, and number on stability of composite panels subjected to compression and shear loadings. This provided a framework for test guidance and for designing a test article capable of showing the real effects of disbonds as it may take place during manufacturing or in-service.

The main objective of the work presented here is to evaluate the benefits of using a robust computational approach to evaluate the effects of composite defects as they may exist in aircraft structural components. Two types of defects were modeled and their effects evaluated: gap and initial delamination. The work is motivated by the desire to reduce testing during the certification of the aircraft.

For gap modeling, two approaches were developed: (1) The first is a mathematical based and uses the wave geometry to determine knockdown in strength and stiffness, and (2) The second is finite element based and models a cross section of the laminate with defects to generate lamina mechanical properties inclusive of the gap defect. The second approach, a high fidelity one, is more generalized and more effective as it can be deployed to evaluate any structure where the defect may exist regardless of the complexity of the structure. The high fidelity approach will be the focus of the applications presented in this paper for assessing effects of gap defects. For modeling initial delaminations, disbonds were included in the finite element models. Then the panels were analyzed using nonlinear buckling analysis to determine the effects of disbonds on stability response.

Next, the analytical procedure used for defect modeling is discussed. It is followed by a description of the method for multi-scale multiphysics durability and damage tolerance analysis. Multi-scale analysis was used to evaluate the strength of stiffened panels. Results from open hole laminates with gap defects are validated with test. Pristine composite stiffened panels are then evaluated for stability and durability. The panels were subjected, respectively, to axial and shear loading. The same panels were also evaluated with gap defects.

3 Description of Analysis Approach
3.1 Modeling of Gaps in Laminated Composites

The approach described here produces lamina properties that are degraded due to the presence of gap defects. This test validated approach is capable of evaluating the effects of gaps regardless of the complexity of the structure (coupon, element, sub-component and component). A computer code was developed to automatically generate lamina properties inclusive of the gap size while an identifier was used to keep track of the lamina that were affected by the gaps to assign them the degraded properties. A brief description of the technical approach is given here:

- Identify the characteristics of the gap by determining whether or not the gap consists of wavy fibers only or combination of wavy fibers and resin rich area as illustrated in Fig. 3.
- Define wave amplitude and gap length from a micrograph of the laminate.
- Generate a three dimensional finite element model for analysis using multi-scale progressive failure analysis MS-PFA [5-7] based on the wave geometry. The fiber is sliced into
- MS-PFA determined the ply properties with the gap after subjecting the finite element model to nine different loading conditions: longitudinal tensile (LT) and compression (LC), transverse tension (TT) and compression (TC), in-plane shear (IPS), out-of-plane shear (13 and 23 directions), out-of-plane tension (33T) and compression (33C).
analysis is briefly described next followed by discussion of results. Fig. 4 shows a two dimensional unit cell cross-section representative of the gap which would be modeled with finite elements and subjected to loading in different directions (in-plane and out-of-plane). This would result in deriving degraded stiffness and strength properties for the ply containing the gap. Next, the multi-scale progressive failure analysis is briefly described followed by validation results of the analytical approach for assessment of manufacturing defects.

3.2 Description of Multi-Scale Progressive Failure Analysis (MS-PFA)

MS-PFA is used to derive mechanical properties for the lamina containing gap defect. As described in the previous section, a finite element model depicting the geometry of the gap is loaded in different directions to calculate lamina stiffness and strength and strain limits. Integrates the following capabilities to enable life prediction of large scale aircraft structures: (a) finite element structural analysis, (b) micro-mechanics, and fracture mechanics options, (c) damage progression tracking, (d) probabilistic risk assessment, (e) minimum damage design optimization, and (f) material characterization codes to scale up the effects of manufacturing defects and local damage mechanisms to the structure level to evaluate overall structural performance and integrity. It takes a full-scale finite element model and accounts for the average material failure at the microscopic level. This capability is integrated into the GENOA software, which was used to perform the analysis presented in the paper.

As shown in Fig. 5, material properties are updated for all iterations, reflecting any changes resulting from damage or crack propagation. MS-PFA’s hierarchical approach allows integration of a wide range of specialized programs, from micro to macro, into an existing verified progressive failure [5-7]. This makes it possible to accomplish synthesis of a variety of composite materials and structures based on progressive failure analysis and virtual testing to predict structure/component safety based on the physics and micro/macro mechanics of materials, manufacturing processes, available data, and service environments. This approach takes progressive damage and fracture processes into account and accurately assesses reliability and durability by predicting failure initiation and progression based on constituent material properties. Physics based failure mechanisms are interrogated at all stages of the loading process. These failure criteria are given in Table 1. The first 9 criteria are stress limits computed by the micromechanical equations based on a material’s constituent stiffness and strength values. Interply delamination due to relative rotation of plies and modified distortion energy (MDE) failure criterion that takes into account combined stresses are also considered. If a failure criterion indicates failure of a lamina, then the mathematically modeled properties of the lamina are changed according to a mathematical sub-model of degradation of the affected material properties. Material and structural equilibrium are achieved before the load is incremented. The analysis process is repeated until catastrophic failure of the structure is detected.

4 Results and Discussion

4.1 Validation of Laminate Response with Gap Defects

Sawicki and Minguet [2] results for the effects of gap defects on laminate compression strength were reproduced analytically using the developed computational approach. Notched and un-notched laminates made from carbon fiber reinforced polymer IM6/3501-6 material with 66% fiber volume and 2% void were evaluated; The notched laminate layup was (45/-45/45/0/90/45/0/90/0/90/0/45/0/90/0/45/45). The open hole compression strength from test and simulation are plotted versus the gap size in Fig. 6. Fig. 7 shows the finite element model used in the open hole compression analysis with MS-PFA. The considered gap size was 2.54 mm. The region where the ply existed used reduced properties reflecting the effect of the gap. Damage initiation was caused by fiber micro-buckling and by shear. The reduction in strength with gap can be traced to the waviness that results in the fibers. The degree of waviness is driven by the height of the gap that depends on the tape thickness and not the gap length. Once the knockdown factor has peaked, increasing gap length does not cause any further increase in knockdown factor (stress concentration is reduced).

Variations in the void content, fiber content, fiber waviness, and fiber micro-buckling resistance were evaluated using sensitivity analysis. It ranked the influence of the considered variables on the open hole laminate failure stress. As illustrated in Fig. 8, fiber-micro-buckling followed by fiber waviness and void content are the most dominant uncertainties that produce variability in failure stress of the open hole laminate. The sensitivity analysis
information can be used as a guide to developing a test matrix. Next, the effect of gaps on flat and stiffened panels are evaluated.

4.2 Effect of Gaps on Linear Buckling Response of a Flat Composite Panel
The objective here is to determine the knockdown factors to be applied on the stability margin of safety to take into account effect of gaps between the tows in the application of flat composite panels. Hillburger and Stearns [8] evaluated the effects of imperfections on buckling response of compression loaded shells. They also evaluated manufacturing flaws in the form of gaps between adjacent pieces of graphite-epoxy tape in laminated plies. Their study determined significant reduction in buckling load could take place when imperfections were present. Although the results presented in this paper agree with the findings reported in reference [8]. The analytical approach for gap modeling made it possible to quickly and reliably evaluate the effects of gaps on overall stability. A flat panel loaded in compression, simply supported on the longitudinal edges and clamped on the smallest edges was used in the stability analysis with and without gaps (Fig. 8). Several gap widths are analyzed 0.762 mm, 3.175 mm, 6.35 mm and 7.62 mm with different patterns and densities. Finite element model and the boundary condition used for this study and a typical first buckling mode shape are shown in Fig. 9.

For evaluation of effects of gap defects on buckling response it is not necessary to model each gap discretely. Knockdown in stability was determined by rerunning the analysis with gap properties defined for the plies with gaps. Effectively the analysis created a region of low modulus fibers which was repeated every time that there is a gap in the adjacent layer following a controlled pattern. Fig. 9 shows a reduction in stability between 10 and 15% for larger gap. The buckling results presented here were based on linear analysis and assuming that gaps existed in every ply resulting in a conservative prediction with respect to gap effects on buckling response.

4.3 Effect of Gaps on Stability and Strength of Stiffened Composite Panels
Stability and strength response of stiffened panels subjected to axial compression and shear loadings are evaluated with and without gap defects. The analysis results presented in this section were obtained using nonlinear static analysis for stability evaluation without invocation of damage tracking. For strength evaluation, multi-scale progressive failure analysis was used. The analysis is repeated for stability and durability (strength evaluation) for two types of panels: pristine and non-pristine containing gap defects. The panels were evaluated two loading conditions: axial compression and shear.

The methodology for evaluating post-buckling behavior of stiffened panels required a three steps approach:
- Perform linear buckling analysis to obtain some skin imperfections to kick start FEM nonlinear solution using classical eigenvalue analysis.
- Infuse some imperfections from linear buckling analysis into the skin geometry and perform nonlinear static analysis (large displacement type solution). Skin and stringer deformations, especially in the out-of-plane directions can be used to detect possible mode jumping (change in stability of panel).
- Repeat non-linear buckling analysis with the pre-shape with MS-PFA. This step involves multi-scale progressive failure analysis with damage tracking and fracture. Load incrementation is carried out until the ultimate load is reached. Fig. 10 shows a flow chart of the overall process used for post-buckling analysis.

The technical approach discussed here was applied for the evaluation of structural stability and strength of stiffened panels under compression and shear loadings. The pristine stiffened panel was loaded once in axial compression and once in shear. Then it was evaluated with large displacement and nonlinear finite element analysis. The load was gradually increased as the geometry was updated by the finite element solver. Material damage evaluation was not included in this analysis. The objective of the analysis was to correlate structural deformations with physical buckling events. The finite element model evaluated contained some imperfections from predefined modes obtained from linear eigenvalue analysis. When the maximum load was reached, the panel deformation indicated presence of buckling cells that were expected for the considered panels and loadings. The out-of-plane displacements at the peak load for the compression and shear panels are presented in Fig. 11.

In addition to the stability analysis, the pristine panels were evaluated for strength using multi-scale progressive failure analysis, which used as input to the analysis a set of fiber and matrix properties that were derived from lamina and laminate test data. The use of fiber and matrix constituent properties enables the evaluation of damage events at their inception source that is the fiber and the matrix [6]. MS-PFA captured all stages of damage evolution including initiation, propagation and ultimate failure. Damage initiation pertains to first ply failure event. Damage propagation
takes place when additional failure criteria are invoked at increased loading. Fig. 12 shows the overall state of damage at ultimate load along with contributing failure mechanisms for the axial compression the pristine panel. Fig. 13 shows the ply-by-ply state of damage in the skin and in the stiffener at the ultimate load for the same loading condition. The simulation opened the way to monitor damage growth event not possible in a test setting. It provides an opportunity to improve the performance of critical structural components if design requirements are not met.

Table 2 summarizes the results obtained for the stiffened pristine panels from stability and strength analysis. Assuming a maximum load for each case of 1.0, the correspondent load ratio for major event is listed in the same table. If structural fracture takes place at a given load, skin buckling is expected to take occurs at about 50% of the fracture load. At fracture, multiple failure modes are active including fiber crushing, fiber micro-buckling, matrix cracking in tension, shear, and delamination.

The stiffened panels were evaluated for stability and strength using various gap defects. Fig. 14 shows the reduction in stability and strength as function of the gap size for the panel loaded in axial compression. Similarly, Fig. 15 shows the reduction in stability and strength as function of the gap size for the panel loaded in shear. The overall reduction in buckling load and strength for the compression panel was below 10% for the maximum gap size considered. The maximum reduction in stability and strength for the shear panel was about 5%. The analysis assumed that the gaps existed in each ply of the evaluated panels. Degraded properties were used in the sections where the gaps were assumed to exist. The panels were tolerant to gap defects due to their redundant load carrying capability and load redistribution after damage initiation. In general, one can conclude the effects of gap defects are limited, especially for the type of gap patterns evaluated.

4.4 Effect of Initial Delaminations on Stability of Composite Panels

Preventing and detecting delamination in composites is a key issue for maintaining reliability of aircraft structures. Because of the low through-the-thickness strength of composites, one primary failure mode has been delamination [10]. Delamination can occur because of material geometrical discontinuities and because impact during maintenance or repair. Effects of initial delamination on the stability of flat composite panels are evaluated for structures subjected to axial compression and shear loading. The objective here is to determine axial compression and shear buckling knockdown factors for various size delaminations and through-thickness position. Finite element models used for this evaluation are shown in Fig. 16 for both axial and shear buckling cases. In the test specimen, delaminations were simulated with two layers of release film giving a nominal thickness of 0.0508mm for the stack-up located at mid thickness. Two rows of nodes going through mid-planes of delamination pieces were created for the two rows of shell elements on the either side of delamination. A cross-section of this geometry is shown in Fig. 17.

The delaminations shown in Fig. 16 as dark areas in the models were varied in size to generate enough points for creation of the knockdown factors for up to a 127mm delamination. Analysis showed that the buckling load of compression panels with 25.4mm delamination was not affected by the presence of the upper right delamination. Furthermore, for sizes greater than 25.4mm it extends under the antibuckling bars and will intersect the other delamination in the center of the panel, therefore, this delamination was dropped from the analysis. Note that the delamination has a squared pattern.

Fig. 18 shows the validation of the stability analysis for flat composite panels loaded in compression. Micro-strains on the front and back side of the panel are compared to test as shown in the figure. The simulation results were obtained from nonlinear buckling analysis. With confidence established in the analysis process, the effects of initial delaminations were evaluated using the configuration shown in Fig. 16 and Fig. 17.

Fig. 19 shows the buckling mode shape obtained with two different delaminating sizes (38.1mm and 127mm) for composite panels loaded in axial compression. The analysis results indicated that small delamination sizes had no effects on the stability of the panel. On the other hand, larger delamination sizes could adversely affect the stability of the panel because of the 35% knockdown factor. Results for the shear panel are presented in Fig. 20 for two delamination sizes (25.4mm and 127mm). For large delamination size, the reduction in buckling strength was greater for composite panels loaded in shear as compared to those loaded in axial compression. Shear loaded panels showed up to 50% reduction in stability for large delamination. This was caused by higher interlaminar stress for shear panels. Many safeguards are in-place to reduce the potential of reduction in stability to initial delamination. These safeguards include design features such as stiffeners, robust design that is insensitive to delamination initiation, materials with enhanced
interlaminar strength, well controlled manufacturing process, and local protection in highly susceptible invisible impact area.

5 Conclusions
Automated fiber placement manufacturing process can inadvertently produce gap defects of varying length that are caused by missing tows. When an over-ply is placed over a ply whose fibers run at a different angle and which has a missing tow, waviness is introduced into the over-ply at the missing tow. It was demonstrated that advanced computational technology integrating micro-mechanics with finite element analysis, damage and fracture tracking, and defect modeling is an effective mean for mitigating risk associated with such defects. At the coupon level, it was concluded that further reductions in compression strength did not occur as the gap size was increased beyond a critical size.

Reduction in compression strength due to the presence of gaps can be traced to the waviness in the fibers. The degree of waviness is driven by the height of the gap that depends on the tape thickness and not the gap length. Once the knockdown factor has peaked, increasing gap length does not cause any further increase in knockdown factor (stress concentration is reduced). The analysis showed that for the stiffened panels, due to redundant load carrying capability and load redistribution, the knockdown in strength and stability was lower than that of the coupon. Simulations presented will be test validated to ensure all knock-down are correlated. The methodology presented allow a reduce test plan definition by exploring different gaps patterns at a simulation level.

The effect of initial delaminations on stability of composite panels was evaluated for panels loaded in axial compression and shear. Validation of the buckling simulation with test results established confidence in the analysis approach. For a 127mm x 127mm initial delamination in a flat panel, it was found that the buckling load could be reduced by 35% for compression and 50% for shear. However, for delamination size of 25.4mm and 50.8mm no loss in buckling strength was expected. The study showed that delamination places a considerable penalty on buckling of aircraft panel structures. However, the stability of stiffened panels is not dependent on the stability of the skin. Therefore, the risk associated with initial delamination is significantly reduced.

6 Acknowledgements
The views and conclusions contained in this article should not be interpreted as representing the official policies, either expressed or implied, of the Bombardier Aerospace.

Fig.1. Typical Automated Fiber Placement gap density in high curvature surface

Fig.2. Automated Fiber Placement 3.175 mm gaps visible on cured part

Fig.3. Gap consisting of wavy fibers or combination of wavy fibers and resin rich area

Fig.4. 2D Unit cell cross-section representative of the ply with gap is modeled with finite element and subjected to loading in different directions
Fig. 5. Progressive failure tracks damage at the micro-scale level of the material.

Table 1
Failure modes considered in multi-scale progressive failure analysis

<table>
<thead>
<tr>
<th>Mode</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>Longitudinal</td>
<td>Fiber tensile strength and the fiber volume ratio.</td>
</tr>
<tr>
<td>Tensile</td>
<td></td>
</tr>
<tr>
<td>Longitudinal</td>
<td>1. Rule of mixtures based on fiber compressive strength and fiber volume</td>
</tr>
<tr>
<td>Compressive</td>
<td>ratio, 2. Fiber microbuckling based on matrix shear modulus and fiber</td>
</tr>
<tr>
<td></td>
<td>volume ratio, and 3. Compressive shear failure or kink band formation that</td>
</tr>
<tr>
<td></td>
<td>is mainly based on ply intralaminar shear strength and matrix tensile</td>
</tr>
<tr>
<td></td>
<td>strength.</td>
</tr>
<tr>
<td>Transverse</td>
<td>Matrix modulus, matrix tensile strength, and fiber volume ratio.</td>
</tr>
<tr>
<td>Tensile</td>
<td></td>
</tr>
<tr>
<td>Transverse</td>
<td>Matrix compressive strength, matrix modulus, and fiber volume ratio.</td>
</tr>
<tr>
<td>Compressive</td>
<td></td>
</tr>
<tr>
<td>Normal Tensile</td>
<td>Plies are separating due to normal tension</td>
</tr>
<tr>
<td>Normal Compressive</td>
<td>Due to very high surface pressure</td>
</tr>
<tr>
<td>In Plane Shear</td>
<td>Failure due to in plane shear with reference to laminate coordinates</td>
</tr>
<tr>
<td>Transverse</td>
<td>Shear Failure due to shear stress acting on transverse cross section</td>
</tr>
<tr>
<td>Normal Shear</td>
<td>oriented in normal direction of the ply</td>
</tr>
<tr>
<td>Longitudinal</td>
<td>Shear Failure due to shear stress acting on longitudinal cross section</td>
</tr>
<tr>
<td>Normal Shear</td>
<td>oriented in a normal direction of the ply</td>
</tr>
<tr>
<td>Modified Distortion</td>
<td>Modified from Distortion Energy combined stress failure criteria used</td>
</tr>
<tr>
<td>Energy Criterion</td>
<td>for anisotropic materials</td>
</tr>
<tr>
<td>Relative Rotation</td>
<td>Considers failure if the adjacent plies rotate excessively with respect to</td>
</tr>
<tr>
<td>Criterion</td>
<td>one another</td>
</tr>
</tbody>
</table>

Fig. 6. Predicted strength knockdown factor for open-hole compression laminate compared to test

Fig. 7. Open-hole finite element model with gap and ply damage due to fiber micro-buckling and matrix cracking in shear

Fig. 8. Sensitivity analysis showing influence of manufacturing parameters on laminate response under compression
Fig. 9. Prediction reduction in stability due to gap defect in flat composite panel subjected to compression loading.

Fig. 10. Strategy for nonlinear buckling analysis with pre-shape and damage tracking.

Fig. 11. Maximum displacement from nonlinear analysis of composite stiffened panels subject to axial compression and shear loading.

Fig. 12. Structural damage in stiffened panel subject to axial compression loading.

Fig. 13. Damage in skin and stiffeners of panel under axial compression at peak load from progressive failure analysis caused by fiber crushing, fiber micro-buckling and matrix shear.

Table 2
Chronology of damage and stability of composite stiffened panels in axial compression and shear.

<table>
<thead>
<tr>
<th>Loading</th>
<th>Maximum Load</th>
<th>Event</th>
<th>Failure Mode</th>
</tr>
</thead>
<tbody>
<tr>
<td>Compression</td>
<td>0.35</td>
<td>Skin buckling</td>
<td>Fiber crushing</td>
</tr>
<tr>
<td></td>
<td>0.63</td>
<td>Skin damage initiation</td>
<td>Fracture load</td>
</tr>
<tr>
<td></td>
<td>0.86</td>
<td>Potential stiffener buckling</td>
<td></td>
</tr>
<tr>
<td></td>
<td>0.93</td>
<td>Stiffener damage initiation</td>
<td></td>
</tr>
<tr>
<td></td>
<td>1.00</td>
<td>Fracture load</td>
<td></td>
</tr>
<tr>
<td>Shear</td>
<td>0.55</td>
<td>Skin buckling</td>
<td>Fiber crushing</td>
</tr>
<tr>
<td></td>
<td>0.73</td>
<td>Skin stiffener damage initiation</td>
<td></td>
</tr>
<tr>
<td></td>
<td>1.09</td>
<td>Fracture load</td>
<td>Multiple</td>
</tr>
</tbody>
</table>

Fig. 14. Predicted knockdown as function of gap size in strength and stability in stiffened composite panel subjected to axial compression loading.

Fig. 15. Predicted knockdown in strength and stability as function of gap size in stiffened composite panel subjected to shear loading.
Fig. 16. Finite element models for axial compression and shear buckling with initial delamination

Fig. 17. Cross section at delamination location showing node splitting

Fig. 18. Test/analysis comparison of micro-strains at the center of compression panel

Fig. 19. Buckling mode shape with two delaminations for panel under axial compression with knockdown factor (KDF) in buckling load

Fig. 20. Buckling mode shape with two delaminations for panel under shear loading with knockdown factor (KDF) in buckling load

References


