Use of PFA to determine design methods for composite stiffened panels with discrete damages

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Presented By: Francesco Di Caprio

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Introduction

- Aeronautical composite structures have to be designed in accordance to degradation factor due to temperature and moisture (T&M) and damage induced by low velocity impact damage (BVID), which can occur during the operative life and which will not be ever repaired.

- With these degradation events, the structure has to withstand up to Ultimate Load.

- Generally, all these requirements are satisfied considering reduced Material Design Values, which are very conservatory.

- Using SHM system and a more advanced reliable numerical tool (like PFA) could remove, in the next future, all these conservatisms and so they can lead to a more efficient use of composite materials.
The PFA can be used to develop a new and more efficient design methodologies.

The following main topics will be shown:

- The Nastran PFA capabilities have been investigated by mean of simple test-cases.
- The PFA has been used to determine a method in order to estimate the strength of composite stiffened panel under compressive load conditions with discrete damage maps. The panel has been designed to be critical at strength.

First Ply Failure  
Centre Bay Failure  
Total Failure
Laminated composite structures can develop local failures, such as matrix cracks, fiber breakage and delamination, under normal operating conditions which may contribute to their final failure.

Hence a reliable methodology for predicting the initiation and growth of such failures is essential to evaluate the performances of composite structures.

Simulating the damage initiation and propagation in composite laminate is the main aim of a Progressive Failure Analysis - PFA.

Usually, the term PFA is used in literature to indicate a methodology able to predict the initiation and propagation only of intralaminar damage (matrix cracking and fiber breakage). The delamination onset and growth is not taken into account in a typical PFA.
The key steps of a PFA are:

1. **Failure Detection**: stress-based failure criteria (Hashin, Hoffman, Puck, etc.) are used in order to detect local lamina failure and determine the mode of failure.

2. **Material Degradation** (Damage models) In general three categories of material degradation models can be identified after ply failure: instantaneous unloading, gradual unloading and constant stress.
Damage onset - Failure Detection

Stress or Strain-based failure criteria are used in order to detect local lamina failure and determine the failure mode.

MSC Nastran offers a lot of failure criteria in order to determine the damage onset (Max Stress, Max strain, Hill, Hoffman, Tsai-Wu, Hashin, Puck and User-defined).

For the applications shown in this presentation only the Hashin’s failure criteria has been used.

The following material properties are needed:

\[ X_T = \text{Allowable tensile stress in the fiber direction.} \]
\[ X_C = \text{Allowable compressive stress in the fiber direction.} \]
\[ Y_T = \text{Allowable tensile stress in the matrix direction.} \]
\[ Y_C = \text{Allowable compressive stress in the matrix direction.} \]
\[ Z_T = \text{Allowable tensile stress in the z-direction.} \]
\[ Z_C = \text{Allowable compressive stress in the z-direction.} \]
\[ S_{xy} = \text{Allowable XY stress.} \]
\[ S_{yz} = \text{Allowable YZ stress.} \]
\[ S_{xz} = \text{Allowable XZ stress.} \]

**Hashin’s Failure Criteria**

<table>
<thead>
<tr>
<th>Matrix Failure - Tensile</th>
<th>[ \sigma_{22} &gt; 0 ]</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>[ \left( \frac{\sigma_{22}}{Y_T} \right)^2 + \left( \frac{\sigma_{12}}{S_{12}} \right)^2 + \left( \frac{\sigma_{23}}{S_{23}} \right)^2 = 1 ]</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Matrix Failure - Compression</th>
<th>[ \sigma_{22} &lt; 0 ]</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>[ \left( \frac{\sigma_{22}}{Y_C} \right)^2 + \left( \frac{\sigma_{12}}{S_{12}} \right)^2 + \left( \frac{\sigma_{23}}{S_{23}} \right)^2 = 1 ]</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Fiber Failure - Tensile</th>
<th>[ \sigma_{11} &gt; 0 ]</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>[ \left( \frac{\sigma_{11}}{X_T} \right)^2 + \left( \frac{\sigma_{12}}{S_{12}} \right)^2 + \left( \frac{\sigma_{13}}{S_{13}} \right)^2 = 1 ]</td>
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<th>Fiber Failure - Compression</th>
<th>[ \sigma_{11} &lt; 0 ]</th>
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<tr>
<td></td>
<td>[ \frac{\sigma_{11}}{-X_C} = 1 ]</td>
</tr>
</tbody>
</table>
In general, three categories of material degradation models can be identified: instantaneous unloading, gradual unloading, and constant stress at ply failure.

MSC NASTRAN offers two different degradation models:

1. Gradual selective stiffness degradation;

2. *Immediate selective stiffness degradation*
Damage Model Description

For gradual degradation method the followed relationship are used:

$$\Delta r_i = -(1-e^{1-F})$$

If the immediate degradation is used, the degradation factor is only a constant value. This is done differently for different criteria. Six such reduction factors are stored and uploaded, they are then used for scaling the respective material modulus according to:

$$E_{11}^{\text{new}} = r_1 E_{11}^{\text{orig}}$$
$$E_{22}^{\text{new}} = r_2 E_{22}^{\text{orig}}$$
$$E_{33}^{\text{new}} = r_3 E_{33}^{\text{orig}}$$
$$G_{12}^{\text{new}} = r_4 G_{12}^{\text{orig}}$$
$$G_{23}^{\text{new}} = r_5 G_{23}^{\text{orig}}$$
$$G_{31}^{\text{new}} = r_6 G_{31}^{\text{orig}}$$

For the criteria that distinguish between fiber and matrix failure (Hashin, Puck) there is a more complex coupling between the failure modes.

**Matrix compression factor:**
$$\Delta r_2 = -(1-a_2)(1-e^{1-F_{mc}})$$

**$E_{33}$ reduction from fiber**
$$\Delta r_3 = -(1-a_4)(1-e^{1-F_f})-a_4(1-e^{1-F_m})$$

**Shear stiffness factor:**
$$\Delta r_4 = -(1-a_3)(1-e^{1-F_m})$$

**Shear reduction from fiber**
$$\Delta r_4 = -(1-a_5)(1-e^{1-F_m})-a_5(1-e^{1-F_f})$$
There are several cases in which the damage evolves really progressively.

For these cases using classical analysis based on first ply failure, instead of PFA analysis, can lead to a real underestimate of the strength of the structure.
Compressive Loaded Composite Panel in Post-Buckling regime

A simple test case has been taken from literature: “Progressive Failure Analysis Methodology for Laminated Composite Structures”, NASA/TP-1999-209107, Sleight D.W.

**Ply Thickness:** 0.136 mm;

**Laminate thickness:** 3.26 mm

**Stacking sequence:** $[\pm 45 / 0_2 / \pm 45 / 0_2 / \pm 45 / 0 / 90]_s$

### Material Properties - T300/5208 Material System

<table>
<thead>
<tr>
<th>Property</th>
<th>Symbol</th>
<th>Value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Longitudinal Young's Modulus</td>
<td>$E_{11}$</td>
<td>131000.44 Mpa</td>
<td></td>
</tr>
<tr>
<td>Transverse Young's Modulus</td>
<td>$E_{22}$</td>
<td>13031.10 Mpa</td>
<td></td>
</tr>
<tr>
<td>Poisson's Ratio</td>
<td>$\nu_{12}$</td>
<td>0.38</td>
<td></td>
</tr>
<tr>
<td>In-Plane Shear Modulus</td>
<td>$G_{12}$</td>
<td>6412.13 Mpa</td>
<td></td>
</tr>
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<td>Longitudinal Tensile Strenght</td>
<td>$X_T$</td>
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</tr>
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<td>Longitudinal Compressive Strenght</td>
<td>$X_C$</td>
<td>1137.64 Mpa</td>
<td></td>
</tr>
<tr>
<td>Transverse Tensile Strenght</td>
<td>$Y_T$</td>
<td>80.94 Mpa</td>
<td></td>
</tr>
<tr>
<td>Transverse Compressive Strenght</td>
<td>$Y_C$</td>
<td>188.99 Mpa</td>
<td></td>
</tr>
<tr>
<td>In-Plane Shear Strenght</td>
<td>$T$</td>
<td>68.95 Mpa</td>
<td></td>
</tr>
</tbody>
</table>
Compressive Loaded Composite Panel – Literature Results

**Reaction vs Out-of-Plane Displacement**

- **Christensen’s Criterion**
- **Hashin’s Criterion**
- **Experimental Results**

**Reaction vs End Shortening**

- **Linear Response**
- **Elastic Non Linear (No Damage)**
- **Experimental Results**

**NASA - Sleight results**

<table>
<thead>
<tr>
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<th>Numerical</th>
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<tr>
<td><strong>Buckling Load [N]</strong></td>
<td>44700</td>
<td>46700</td>
</tr>
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<td><strong>Failure Load [N]</strong></td>
<td>97400</td>
<td>107000</td>
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**Experimental Buckling Mode**
Evaluation of the right scale factor for the mesh perturbation

For a scale factor equal to 1, the maximum out-of-plane displacement is equal to 1 mm. Increasing the scale factor the out-of-plane behavior comes up to experimental behavior. High scale factors lead to a wrong stiffness estimate (pert=1)
First Test-Cases Results

**Results Table**

<table>
<thead>
<tr>
<th></th>
<th>Buckling Load [N]</th>
<th>Failure Load [N]</th>
<th>Buckling Error % (Experimental)</th>
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<td>46700</td>
<td>107000</td>
<td>-4.47</td>
<td>-9.86</td>
</tr>
<tr>
<td>MSC Nastran</td>
<td>45470</td>
<td>100581</td>
<td>-1.72</td>
<td>-3.27</td>
</tr>
</tbody>
</table>

Mesh perturbation factor = 0.1; Degradation factor (MATF) = 0.01
First Test-Cases Results

Fiber Damage:
- Non-linear: 78.5067 % of Load, Fiber Damage for Progressive Failure, Maximum 24 of 24 layers
- Deform: SCI Step 1: A1 Non-linear: 100% of Load, Displacements, Translational, (NON-LAYERED)

Matrix Damage:
- Non-linear: 78.5067 % of Load, Matrix Damage for Progressive Failure, Maximum 24 of 24 layers
- Deform: SCI Step 1: A1 Non-linear: 100% of Load, Displacements, Translational, (NON-LAYERED)
**Second Test-Case Description**

**Compressive Loaded Composite Panel with hole in post-buckling regime**

A simple test case has been taken from literature: “Progressive Failure Analysis Methodology for Laminated Composite Structures”, NASA/TP-1999-209107, Sleight D.W.

**Numero Ply :** 24 ;

**Spessore Ply :** 0.1458 mm ; **Spessore Totale** ≈3.5mm

**Stacking sequence:** \([\pm 45 / 0 / 90]_s\)

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Compressive Loaded Composite Panel with hole – Literature Results

**Reaction vs End Shortening**

![Graph showing reaction vs end shortening](image)

**Reaction vs Out-of-Plane Displacement**

![Graph showing reaction vs out-of-plane displacement](image)

**NASA - Sleight results**

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<td>65273</td>
</tr>
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<td><strong>Failure Load [N]</strong></td>
<td>94022</td>
<td>77026</td>
</tr>
</tbody>
</table>

**Experimental Buckling Mode**
Increasing the scale factor the out-of-plane behavior comes up to experimental behavior.

A perturbation factor equal to 0.175 has been chosen in order to obtain a mesh perturbation equal to 5% of total thickness (best correlation).

High scale factors lead to a wrong stiffness estimate (pert=1)
A parametric study has been performed in order to determine the right degradation factor $k$, for *IMMEDIATE* selective stiffness degradation method.

The best correlation has been reached with $k=0.1$.

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**Results Table**

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<th>Failure Load [N]</th>
<th>Buckling Error %</th>
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<tr>
<td>Experimental - Sleight</td>
<td>63465</td>
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<td></td>
<td></td>
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<tr>
<td>Numerical - Sleight</td>
<td>65273</td>
<td>77026</td>
<td>-2.85</td>
<td>18.08</td>
</tr>
<tr>
<td>NASTRAN</td>
<td>64840</td>
<td>89405</td>
<td>-2.17</td>
<td>4.91</td>
</tr>
</tbody>
</table>
A parametric study has been performed in order to determine the right degradation factor $k$, for **GRADUAL** selective stiffness degradation method.

The gradual method overestimates the failure load.
Second Test-Case Results

Out-of-Plane Displacements

- Fringe: SC1; A1 Non-linear: 100% of Load, Displacements, Translational, Z Component, (NON-LAYERED)
  - Max: 9.22x10^3
  - Min: -1.10x10^3

Damaged Elements

- Total Damage for Progressive Failure: Maximum 24 of 24 layers
  - Damaged
  - Undamaged

The aim of the test is the strength reduction evaluation of a composite plate with a hole both by experimental and numerical point of view.

The specimens derives from CAI-TEST specification.

The width specimen has been chosen in order to simulate the behavior of infinite plate with a hole (W/D=16.6)

Layup: [45, -45, 0, 90]_3s
Composite Plate with a hole: Experimental Results

Undamaged

With Hole

In Test

Before

After

In Test

After

Before

After
Composite Plate with a hole: Experimental Results

The ultimate strength of the undamaged plate is equal to 177.5 kN (mean value).
The ultimate strength of the holed plate is equal to 115 kN (mean value).
The factor $K_{\text{notch}}$ is equal to 0.65.

<table>
<thead>
<tr>
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<th>Strength [N]</th>
<th>Failure Strain [µε]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Undamaged</td>
<td>177543.78</td>
<td>7975.72</td>
</tr>
<tr>
<td>With Hole</td>
<td>114922.06</td>
<td>4626.83</td>
</tr>
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</table>

Experimental Results
Composite Plate with a hole: Numerical Results

The numerical factor $K_{\text{notch}}$ is equal to 0.61

<table>
<thead>
<tr>
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<th>Strength [N]</th>
<th>Failure Strain [µε]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Undamaged</td>
<td>183486.08</td>
<td>7500.00</td>
</tr>
<tr>
<td>With Hole</td>
<td>110326.11</td>
<td>4576.00</td>
</tr>
</tbody>
</table>

Numerical Results

Reaction vs. Applied Strain

- UNDAMAGED
- UnDamaged - FPF
- With Hole
- With Hole - FPF

2976 µε

4042 µε

4576 µε

4600 µε
The differences between Experimental and numerical results are quite small (considering also the experimental results variability)

<table>
<thead>
<tr>
<th>Experimental Results</th>
<th></th>
<th></th>
</tr>
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<td></td>
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<td>4576.00</td>
</tr>
</tbody>
</table>

| Experimental-Numerical|  |  |
|                       | **Stength [N]**  | **Failure Strain [με]** |
| UD - Error %          | -3.35            | 5.96     |
| H - Error %           | 4.00             | 1.10     |
**K\textsubscript{notch} determination by OHC virtual tests**

Open Hole Compressive Tests have been performed in order to evaluate the layup influence on the failure strain and on factor \( K\text{notch} \).

Different plate with increasing width have been considered. The plies number is fixed.

- The failure strain values are greatly influenced by layup.
- The intensification stress factor is greatly influenced by layup, this in widely documented in literature.
- The worst layup has been used for the following analyses.
- In the design phase it is mandatory to use a suitable \( K\text{notch} \) and allowables, which can vary with the 0°ply percentage as demonstrated by theoretical activities.

![Knotch vs. % 0°](image)
Application: Stiffened Panel

Composite Stiffened Panel with one hole

The composite stiffened panel under investigation has been sized in order to reach the failure load before the panel buckling load.
**Stiffened Panel: Results**

- **Advanced design with PFA (using undamaged allowables):**
  - With Hole and UDA - First ply Failure load: 792 kN – 1966 με
  - With Hole and UDA - Total Failure load: 1406 kN – 3525 με

- **Classical design (using damaged allowables):**
  - First ply Failure load: 1499 kN – 3757 με
  - Total Failure load: 1499 kN – 3757 με

Due to the load conditions the panel failure load with damaged allowables is quite close to one obtained with undamaged allowables and one hole.
Stiffened Panel: Results

Evolution of damage under compressive load

- First ply Failure load: 792 kN – 1966 με
- Total Failure load: 1406 kN – 3525 με
Stiffened Panel: Conclusions

The ultimate strain of the stiffened panel, designed in accordance to strength criteria, under compressive load and with a hole (in the skin center), can be determined considering the asymptotic value related to OHC test (W/D>>1).

Summary Table

<table>
<thead>
<tr>
<th>Model</th>
<th>Undamaged FSnP [µε]</th>
<th>$K_{\text{notch}}$</th>
<th>Predicted Value FSnP [µε]</th>
<th>Failure Strain (FSnP) [µε]</th>
</tr>
</thead>
<tbody>
<tr>
<td>6 Stringer Panel (Layup2)</td>
<td>5429.79</td>
<td>0.65</td>
<td>3529.36</td>
<td>3525.12</td>
</tr>
<tr>
<td>2 Stringer Panel (Layup2)</td>
<td>5429.79</td>
<td>0.65</td>
<td>3529.36</td>
<td>3411.43</td>
</tr>
<tr>
<td>2 Stringer Panel (Layup QI)</td>
<td>5429.79</td>
<td>0.73</td>
<td>3963.75</td>
<td>3888.65</td>
</tr>
</tbody>
</table>

$K_{\text{notch}} = \frac{\text{OHC}_{\text{layup}}}{\text{Unnotched}}$

$\varepsilon_{\text{stiffened panel without hole}}$

$\varepsilon_{\text{stiffened panel with a hole}} = K_{\text{notch}} \times \varepsilon_{\text{stiffened panel without hole}}$
Future Activities

- At the moment at CIRA we are studying how to simulate in different way the damage induced by low energy impact (BVID), in order to reduce the time related to numerical simulation of CAI test.
- Under certain hypotheses we can model the BVID with a hole with a suitable diameter.
- PFA can support the design in order to evaluate the real strength of the structure, under a realistic damage maps, using full undamaged allowables.
Thank you for your attention