PRELIMINARY DESIGN OF A COMPOSITE MATERIAL WING FOR A GENERAL AVIATION AIRCRAFT

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Supervisors:

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Thesis activity

- The wing structure of an aircraft currently realized in aluminium alloy is redesigned in composite material.

- Layout of the wing structure: based on the original layout to have no variations in the fuselage structural scheme.
- Only the main wing spar passes through the fuselage.
Objectives

• Stiffness equal to or greater than the one of the original wing

• No buckling up to ultimate load

• Weight reduction
Load cases

- Most critical load cases for static analysis of the original wing:
  - Pull-up manoeuvre (maximum load factor) at $V_A$
  - Negative manoeuvre
  - Maximum roll acceleration: sudden deflection of the ailerons at $V_A$
Materials

- Carbon fibre epoxy-matrix composites
- Mechanical properties in elevated temperature wet conditions

<table>
<thead>
<tr>
<th></th>
<th>Unidirectional</th>
<th>Fabric</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Longitudinal</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Tension modulus [MPa]</td>
<td>164000</td>
<td>62000</td>
</tr>
<tr>
<td>Compression modulus [MPa]</td>
<td>142500</td>
<td>62000</td>
</tr>
<tr>
<td><strong>Transverse</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Tension modulus [MPa]</td>
<td>5600</td>
<td>62000</td>
</tr>
<tr>
<td>Compression modulus [MPa]</td>
<td>8100</td>
<td>62000</td>
</tr>
<tr>
<td><strong>Shear</strong></td>
<td></td>
<td></td>
</tr>
<tr>
<td>In-plane modulus [MPa]</td>
<td>2100</td>
<td>2200</td>
</tr>
<tr>
<td>Poisson’s ratio [-]</td>
<td>0.39</td>
<td>0.05</td>
</tr>
<tr>
<td>Density [kg/m³]</td>
<td>1580</td>
<td>1570</td>
</tr>
<tr>
<td>Fiber volume</td>
<td>57.3%</td>
<td>55.5%</td>
</tr>
<tr>
<td>Ply thickness [mm]</td>
<td>0.125</td>
<td>0.28</td>
</tr>
</tbody>
</table>

- Hexagonal NOMEX® honeycomb

1 National Institute for Aviation Research, Wichita State University, 2011
Failure criteria

- Laminate: First Ply Failure
- Ply: Max Strain Failure Criterion:

\[
\begin{align*}
X_{\varepsilon_C} &< \varepsilon_{xx} < X_{\varepsilon_T} \\
Y_{\varepsilon_C} &< \varepsilon_{yy} < Y_{\varepsilon_T} \\
|\gamma_{xy}| &< S_{\gamma_{12}}
\end{align*}
\]

\[X_{\varepsilon_C}, X_{\varepsilon_T}, Y_{\varepsilon_C}, Y_{\varepsilon_T} \text{ and } S_{\gamma_{12}} \text{ are the allowable strains}\]

- Failure index: ratio of applied strain to allowable strain (must be less than 1)
- CAI (Compression After Impact) allowable strains depend on laminate thickness
Failure criteria

- CAI (Compression After Impact) allowable strains depend on laminate thickness
The whole structure is modelled with CQUAD4 or CTRIA3, except for the rib flanges (CROD).

- Overlaps between covers elements and spar caps elements
- Upper and lower covers are realized with sandwich panels, but honeycomb is not present in the spar cap area.
Optimization

• Optimization performed only on main spar caps and covers sandwich panels (the other components of the wing are simply designed with traditional techniques)

• Objective function to be minimized: mass

• Design constraints:
  – Nodal displacements of the wing tip are constrained in the normal direction to the wing plane (bending stiffness)
  – Failure index less than 1 (strength constraint)
  – No buckling up to ultimate load

• Manufacturing constraints
Phase I (free-sizing): determines the concept design of ply shapes and thicknesses: for each super-ply Optistruct calculates 4 ply shapes that constitute the starting model for Phase II.
Optistruct optimization of composite structures (2/2)

- Phase II (sizing): new constraints can be introduced in this phase, that determines the number of plies of each ply patch (phase (b) e (c))
- Phase III (ply-stacking optimization): determines the detailed stacking sequence, considering various ply book rules (phase (d))
Free-sizing optimization (1/4)

- Starting model of the upper cover (same stacking sequence considered for the lower cover):
  - 1 fabric super-ply at a $\pm 45$-degree orientation
  - 1 fabric super-ply at a 0/90-degree orientation
  - 1 unidirectional super-ply at a 0-degree orientation
  - 1 honeycomb super-ply at a 0-degree orientation

- Starting model of the spar caps:
  - 1 fabric super-ply at a $\pm 45$-degree orientation
  - 1 unidirectional super-ply at a 0-degree orientation

- SMEAR formulation for the laminates
Free-sizing optimization (2/4)

- Optimization constraints:
  - Nodal displacements of the wing tip (bending stiffness)
  - No buckling up to ultimate load
  - Four fabric plies minimum on each spar cap
  - Minimum 0.5 mm laminate thickness on both sides of the honeycomb
  - Minimum 6.35 mm honeycomb thickness
  - In this phase constraints on failure indices not allowed
Free-sizing optimization (3/4)

Iteration history of the objective function: big changes in the first few iterations
Free-sizing optimization (4/4)

- Unidirectional plies calculated from the original super-ply on the upper spar cap:

- Unidirectional plies calculated from the original super-ply on the upper cover:
Sizing optimization (1/3)

• Starting model: ply shapes obtained after free-sizing optimization adjusted to be manufacturable

• Unnecessary plies removed (infinitesimal thickness)

• SYM (symmetric) formulation for the laminates

• New constraints added to the previous ones:
  – Failure indices less than 1 everywhere (strength constraints)
  – Allowable strains of a thick laminate (avoid excessive constraints in optimization)
Sizing optimization (2/3)

Final models after free-sizing optimization (figures above) and starting models for sizing optimization (figures below)

Upper spar cap, UD plies

Upper cover, UD plies
Sizing optimization (3/3)

Iteration history of the objective function: to satisfy the strength constraints introduced in this phase, the structure needs a greater mass.
Local patches

• Static analysis performed after optimization considering the correct allowable strains for each area with different thickness
• Some local strength problems on the upper and lower covers (red areas)

• Some ply shapes modified and some local patches placed in critical areas
Optimization results

Upper cover: symmetric sandwich with a 6.35 mm-thick honeycomb
Only the laminate on one side of the core here represented: ply 10 is adjacent to the honeycomb

<table>
<thead>
<tr>
<th>No ply</th>
<th>Colour</th>
<th>Unidirectional/Fabric</th>
<th>Fiber orientation</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Red</td>
<td>Fabric*</td>
<td>±45°</td>
</tr>
<tr>
<td>2</td>
<td>Green</td>
<td>Fabric</td>
<td>±45°</td>
</tr>
<tr>
<td>3</td>
<td>Green</td>
<td>Fabric</td>
<td>±45°</td>
</tr>
<tr>
<td>4</td>
<td>Pink</td>
<td>Fabric</td>
<td>±45°</td>
</tr>
<tr>
<td>5</td>
<td>Pink</td>
<td>Fabric</td>
<td>±45°</td>
</tr>
<tr>
<td>6</td>
<td>Blue</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>7</td>
<td>Yellow</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>8</td>
<td>Yellow</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>9</td>
<td>Purple</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>10</td>
<td>Red</td>
<td>Fabric</td>
<td>0°/90°</td>
</tr>
</tbody>
</table>

* Red plies are placed on the whole surface
Optimization results

Lower cover: symmetric sandwich with a 6.35 mm-thick honeycomb
Only the laminate on one side of the core here represented: ply 8 is adjacent to the honeycomb

<table>
<thead>
<tr>
<th>N° ply</th>
<th>Colour</th>
<th>Unidirectional/Fabric</th>
<th>Fiber orientation</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Red</td>
<td>Fabric*</td>
<td>±45°</td>
</tr>
<tr>
<td>2</td>
<td>Yellow</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>3</td>
<td>Yellow</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>4</td>
<td>Blue</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>5</td>
<td>Blue</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>6</td>
<td>Green</td>
<td>Fabric</td>
<td>0°/90°</td>
</tr>
<tr>
<td>7</td>
<td>Red</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
<tr>
<td>8</td>
<td>Red</td>
<td>Unidirectional</td>
<td>0°</td>
</tr>
</tbody>
</table>

* Red plies are placed on the whole surface
Second optimization

- New optimization performed only on spar caps, laminates of upper and lower covers are fixed
- Some plies can be removed thanks to the patches on the covers
- Free-sizing optimization gives ply shapes that are manually adjusted

Upper spar cap, UD plies

Lower spar cap, UD plies
Optimization results

Spar caps

Thickness [\text{\text{-\text{}}}] vs. Wingspan [\text{mm}]

- Upper cap
- Lower cap
Optimization results

- Buckling load: 117% of the ultimate load
Ply-stacking optimization

- Unnecessary in this case. Simple considerations determine the stacking sequence shown:
  - The laminates are symmetric and the core is at the center of the sequence
  - The plies placed on the whole surface must be the cover layers of the solid laminate on one side of the honeycomb
  - Local patches are at the center of the sequence of the solid laminate on one side of the core
Real structure mass estimation

- Various elements not present in the finite element model:
  - Lightning strike protection
  - Hysol® Synskin® HC 9837.1™
  - Primer
  - Structural adhesives
  - Supported adhesive for sandwich panels fabrication
  - Bolts and rivets
  - Quasi-isotropic laminates for mechanical fasteners
  - Symmetric spars
  - Mass increase due to project development, strength problems in structural tests, unexpected manufacturability constraints

Composite materials allow a mass reduction of **25%** of the structural mass of the original wing box.
Conclusions

• The study of a composite materials alternative for a wing box structure led to a significant weight reduction

• All relevant load conditions and constraints must be considered in structural optimization

• Excessive constraints must be avoided not to cause a useless mass increase
Suggestions for future work

• Project development to a more detailed level

• Verify the robustness of the optimization procedure: optimization of the whole structure

• Take a deeper look into the fabrication technologies required (concurrent engineering)

• Study of a different structural scheme with the wing box passing through the fuselage
THANKS FOR YOUR ATTENTION