Multi-level integrated structural sizing of a composite sandwich wing

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Abstract

In this work a procedure for the structural sizing of a composite sandwich wing is described. The procedure is named MLISS (Multi-Level Integrated Structural Sizing); it foresees three growing levels of detailed design and integrates the features of different analysis tools: in-house codes written in Matlab and Fortran languages, a commercial structural sizing software and a finite element software.

The wing skin is devised in composite honeycomb-core sandwich panels while the spar webs and the caps in solid laminate; MLISS sizes the structure using the external loads determined by an in-house code developed for the aeroelastic analyses (AEDC).

In the first level, the wing is considered as concentrated elements: the skin is sized by bending and torsion; the spar webs and the caps by torsion and bending respectively. The skin carries only the in-plane and bending loads; no transverse shear effect is considered (the core thickness is zero): the skin panels are simulated in solid laminate of thickness two times the thickness of a single sandwich face. Elementary theories are applied to determine the internal loads acting on the different structural wing components. In the second level, the internal loads are used to update the wing sizing. The contribution of the honeycomb core for the skin panels is now considered and its optimal thickness is evaluated. In the third level the finite element model is built; the internal FEA loads are evaluated to size definitively the structure by an iterative approach. Applied to a high aspect ratio wing, MLISS has produced satisfying results; moreover the structural weight obtained in the first level is not far from that one obtained in the third level: so for a fast preliminary weight estimation of a composite sandwich wing, only the first sizing level can be applied.

Keywords: wing, structural sizing, honeycomb-core sandwich panels, multi-level, integrated, composite material.
1 Introduction

In the last years the development of multi-level approaches for the structural optimization of aerospace structures in composite material is awakening interest very much in the fields of the research and of the industry (Carrera et al. [1], Romano et al. [2], Gasbarri et al. [3], Liu et al. [4]).

The present work deals with the implementation of a multi-level and integrated procedure named MLISS, Multi-Level Integrated Structural Sizing, finalized to the structural sizing of a wing in composite material.

The procedure is multi-level because characterized by three different levels of growing detailed design, and integrated because it integrates different structural sizing tools: in-house developed and commercial codes. This paper describes the different levels of this procedure and the theoretical assumptions to the base of it.

MLISS sizes a wing so devised: the skin is in composite sandwich with core in honeycomb while the facings are carbon-epoxy laminates; the spars and the ribs are in solid laminate of the same carbon-epoxy material of the skin facings.

In this paper the procedure is applied to a high aspect ratio wing, typical of long endurance and high altitude unmanned aerial vehicles. The external sizing loads are given by an in-house developed code for the aeroelastic analyses (AEDC). The structural sizing is executed by means of an iterative approach AEDC/MLISS starting from the loads evaluated on the rigid structure.

2 Wing description and materials

The wing is characterized by a high aspect ratio of 30.5, with a span of 61 m and an area of 122 m²; the root and the tip chords are 4.0 m and 1.0 m respectively. In fig. 1 an image of the right semi-wing is shown. The wing has a bi-spar configuration: the front and rear spar locations are 25% and 60% chordwise respectively. The wing box configuration is mono-cell from \( y = 4.0 \) m to \( y = 6.0 \) m (fig. 2) and bi-cell from \( y = 6.0 \) m to \( y = 30.5 \) m (fig. 3).

![Figure 1: Right semi-wing.](image-url)
The honeycomb core material is **HRH-10-1/8-1.8 (Aramid Fiber/Phenolic Resin Honeycomb)**; the property values are shown in tab. 1. The minimum and the maximum core thicknesses available are **5.08 mm** and **914.4 mm** respectively.

<table>
<thead>
<tr>
<th>Property</th>
<th>Value 1</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>density</td>
<td>29</td>
<td>Kg/m³</td>
</tr>
<tr>
<td>$E_s$</td>
<td>0.055</td>
<td>Gpa</td>
</tr>
<tr>
<td>$E_c$</td>
<td>0.055</td>
<td>Gpa</td>
</tr>
<tr>
<td>$G_w$</td>
<td>0.01</td>
<td>Gpa</td>
</tr>
<tr>
<td>$G_{11}$</td>
<td>0.026</td>
<td>Gpa</td>
</tr>
<tr>
<td>$F_{tu}$</td>
<td>0.69</td>
<td>Mpa</td>
</tr>
<tr>
<td>$F_{cu}$</td>
<td>0.69</td>
<td>Mpa</td>
</tr>
<tr>
<td>$F_{sug}$</td>
<td>0.31</td>
<td>Mpa</td>
</tr>
<tr>
<td>$F_{rad}$</td>
<td>0.55</td>
<td>Mpa</td>
</tr>
<tr>
<td>Cell size</td>
<td>3.175</td>
<td>mm</td>
</tr>
</tbody>
</table>

The honeycomb cell type is hexagonal (fig. 4): this kind of shape gives minimum density for a given amount of material respect to rectangular cells that increase shear proprieties in $W$ direction and, reduce lightly the same properties in the $L$ direction how much it happens as regards for hexagonal cells. The core ribbon direction is assumed equal to the wing span direction.

Table 2 shows the property values of the carbon-epoxy material (**prepreg**), **IM7/977-2**, used for the facings of the skin, the spars and the ribs: a knockdown
factor (of about 50%) has been applied to the strength properties to take in account moisture and impact damage effects.

![Hexagonal cell type [5].](image)

Figure 4: Hexagonal cell type [5].

Table 2: Carbon-epoxy material properties.

<table>
<thead>
<tr>
<th>Property</th>
<th>Value 1</th>
<th>Value 2</th>
</tr>
</thead>
<tbody>
<tr>
<td>Density</td>
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<tr>
<td>$E_{2}'$</td>
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<td>GPa</td>
</tr>
<tr>
<td>$E_{2}^c$</td>
<td>9.31</td>
<td>GPa</td>
</tr>
<tr>
<td>$F_{2}^{tu}$</td>
<td>930.83</td>
<td>MPa</td>
</tr>
<tr>
<td>$F_{2}^{cu}$</td>
<td>930.83</td>
<td>MPa</td>
</tr>
<tr>
<td>$\varepsilon_{2}^{tu}$</td>
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<td></td>
</tr>
<tr>
<td>$\varepsilon_{2}^{cu}$</td>
<td>0.1</td>
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<tr>
<td>$E_{1}'$</td>
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<td>GPa</td>
</tr>
<tr>
<td>$E_{1}^c$</td>
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<td>GPa</td>
</tr>
<tr>
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<td>MPa</td>
</tr>
<tr>
<td>$F_{1}^{cu}$</td>
<td>592.97</td>
<td>Mpa</td>
</tr>
<tr>
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<td>GPa</td>
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<td>$\gamma_{12}$</td>
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<td>$v_{12}$</td>
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<tr>
<td>Ply thickness</td>
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<td>mm</td>
</tr>
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</table>

3 MLISS procedure

The MLISS procedure foresees three different levels of growing detailed design, as it is shown in fig. 5.

For the structural sizing only symmetric and balanced laminates are considered.
Figure 5: MLISS procedure.

The wing is sized at ultimate loads according to:
- no instability/no strength failure;
- minimum weight requirement.

In particular, for the honeycomb core sandwich skin, its typical instabilities are considered: core shear crimping, face wrinkling instability, face dimpling.

The wing is considered as subdivided in more characteristic sections along the wingspan; each section is composed of more bays (subpart between two consecutive ribs). For each section the more load carried panels of the skin and of the spar webs are considered and optimized; such panels size the whole section. Except for the wing geometry, the unique input file of the procedure is that one containing the external loads acting in assigned wing cross sections (coincident with the rib cross sections).
3.1 First level

Two codes, in Matlab language, have been developed: Dim_Wing_Box.m and Stress_Section.m. The first one sizes the structure while the second one evaluates the internal loads.

In Dim_Wing_Box.m, the wing box is considered as concentrated elements. For simplicity, the laminates are considered characterized by only plies at ±45° (always symmetric and balanced); in this case, the laminate Young modules \( E_1 \) and \( E_2 \) are equal.

The spar caps are simulated by unidirectional laminates: the fibers are oriented in the wing span direction. They are sized using the rate of bending moment acting in each wing cross-section; it’s assumed that the two spars withstand the same normal stress. If \( M_{B,i} \) is the bending moment acting in the generic cross section \( i \), \( I_i \) the inertia moment, \( h_i \) the spar height, the rate of bending moment acting on each spar is:

\[
M_{i,F}^{\text{spar}} = h_{i,F}^2 \left( \frac{h_i}{I_i} \right) M_{B,i}, \quad M_{i,R}^{\text{spar}} = h_{i,R}^2 \left( \frac{h_i}{I_i} \right) M_{B,i}
\]

with

\[
h_i = \frac{1}{I_i \left( \frac{1}{I_{i,F}} + \frac{1}{I_{i,R}} \right)}
\]

where the subscriptions \( F \) and \( R \) are relative to the front spar and rear spar respectively. The spar cap areas are sized at strength:

\[
A_{i,F}^{\text{cap}} = \frac{M_{i,F}^{\text{spar}}}{F_{1,F} h_{i,F}}, \quad A_{i,R}^{\text{cap}} = \frac{M_{i,R}^{\text{spar}}}{F_{1,R} h_{i,R}}
\]

these values are referred to the single cap of the relative spar.

The skin is sized at torsion and bending. In this level the honeycomb core of the skin is not considered; the skin is simulated as an equivalent solid laminate of thickness two times the hypothetic thickness of a single sandwich facing, as shown in fig. 6. In this way only the in-plane and bending behaviour of the sandwich panel is considered, no transverse-shear effects are taken in account. In fact, for honeycomb sandwiches it’s possible to assume that all of the in-plane and bending loads are carried by the facings only, while the core resists transverse shear loads, increases the stiffness of the structure by holding the facing skins apart and gives support to the facing skins to produce a uniformly stiffened panel ([6]).

In the generic cross section \( i \), the elastic axes is given by the sequent formula:

\[
x_i \approx \frac{\left( \frac{h_{i,R}}{h_{i,F}} \right)^3}{1 + \left( \frac{h_{i,R}}{h_{i,F}} \right)^3}
\]
where $x_i$ is measured from the front spar, $d_i$ is the distance between the two spars. For the skin sizing the pure shear fluxes are neglected (iso-static fluxes); only the torsional fluxes (iper-static fluxes) are considered using Bredt formula. Referring to the more complex bi-cell configuration, fig. 3, if $M_{T,i}$ is the total torsion acting in the cross section $i$, imposing the equilibrium and the consistency equations, each cell withstands the following contribution:

$$M_{t,\text{cell1},i} = \frac{J_{\text{cell1},i}}{1 + J_{\text{cell1},i} / J_{\text{cell2},i}} M_{T,i} \quad \text{and} \quad M_{t,\text{cell2},i} = M_{T,i} - M_{t,\text{cell1},i}$$

where $J_{\text{cell}}$ is the torsional cell constant for thickness unity; the subscription $t$ is for torsional. Sizing the skin at strength, the thickness of the generic cell is given by

$$t_{\text{skin},i} = \frac{M_{t,\text{cell},i}}{2 \Omega_{\text{cell},i} F_{\text{su}}}.$$ (1)

Figure 6: Cross-section of sandwich and monocoque construction [5].

The bending contribution of the skin between the two spars is simulated considered two dummy areas, on the upper and lower skin; they represent the skin area contribution to the bending withstanding (fig. 7).

In the generic wing cross section $i$, referring to fig. 7, the following empirical formula is assumed for the two areas (the subscription $b$ is for bending):

$$A_{\text{skin},b,\text{upper},i} = \frac{1}{3} \sum_{j=1}^{2} A_{\text{cap, upper},j}, \quad A_{\text{skin},b,\text{lower},i} = \frac{1}{3} \sum_{j=1}^{2} A_{\text{cap, lower},j}.$$ 

These areas are spread on the skin, upper and lower respectively; the final skin thickness, for a more conservative approach, is the maximum thickness among the two ones (upper and lower) determined from the above areas.

The total skin thickness $t_{\text{skin},i}$ in the generic cross section $i$, is equal to the sum of the two thickness due to the torsion and bending respectively:

$$t_{\text{skin},i} = t_{t,i}^{\text{skin}} + t_{b,i}^{\text{skin}}.$$
For the leading edge skin, only the torsional contribution is considered. The spar webs are sized by torsion using the eqn. (1).

Finally, for the wing so-sized, the code Stress_Section.m evaluates, by elementary theories, the internal loads on each structural component and the weight structure too.

### 3.2 Second level

The internal loads evaluated in the previous level are used in this level to optimize the stacking sequence of the spar webs, skin facings and to size the skin honeycomb core thickness, which, in this level, is not neglected; the transverse shear effect is taken in account. This optimization is developed using structural sizing software Hypersizer (Collier Research ver. 3.7.1) by means of a Free Body Diagram (FBD) approach ([7]). In this approach Hypersizer uses manual input of loads (Stand Alone approach).

This software analyzes and optimizes structural components, which are pieces of panels and beams. Based on the panel length, width, concept, shape, material and layups, Hypersizer computes the corresponding virtual loads that bring the panel into FBD force equilibrium. The external loads are resolved internally into stress resultants on each analysis object such as flanges, webs, etc. The FBD state of internal stresses and strains for all panel segments are integrated and summed to verify equilibrium of forces and strain compatibility for the panel/beam as a whole. The panels/beans are evaluated for strength and stability for the applied loads and boundary conditions using analysis approaches based on traditional industry methods, modern analytical and computational solutions.

In this case the wing skin is in honeycomb sandwich and the wing spar webs in solid laminate; assigned the composite material properties, the layup optimization and the core thickness sizing of these components is achieved, considering a large number of candidate layup families (also than more 400), with all strength and stability margin of safety (\(MS = \frac{P_{\text{critical}}}{P_{\text{acting}}} - 1\)) positive according to the minimum weight requirement.

### 3.3 Third level

In this step the wing finite element model is built satisfying the layups and the core thicknesses optimized in the previous step: the skin and the web panels are
2-D elements while the spar caps rod elements. The FEA internal loads acting on the different structural components of the wing are evaluated using MSC Nastran software ([8]).

These loads are the starting point of an iterative procedure FEA/Hypersizer: FEA internal loads evaluation/structural optimization.

In this case the finite element model (geometry and internal loads) is imported in Hypersizer; using FEA computed internal loads Hypersizer permits the automation of the full design, analysis and build process ([9], [10]). The finite element method is used to resolve the general boundary conditions and the applied external loads into internal loads that are used to size the panels and the beams. Hypersizer uses statistical methods to determine the appropriate design-to-loads acting on each structural component.

For the different wing items, the best layups and core thicknesses satisfying the minimum weight requirement are selected, with all strength and stability margins of safety positive. Just to summarize, in this approach there are times for

- reading FEA internal loads,
- performing the optimization,

and

- updating the FEM model.

The finite element model properties are updated by means of an in-house developed code that updates the Nastran cards automatically.

The iterative procedure FEA/Hypersizer runs until a convergent solution is obtained.

4 Results and conclusions

The paper has proposed an integrated and multi-level procedure for the structural sizing of a wing in composite material with the skin in honeycomb core sandwich.

The reference wing sized in this work has a high aspect ratio ($\geq 2$); the final weight of this wing, sized with the MLISS procedure, is resulted $908$ Kg, against $934$ Kg estimated by means of a preliminary (conceptual) design. This result comforts the use of such procedure. Another important result is that the weight obtained in the first level is lightly different from that one obtained by the execution of the entire three levels; so, for a preliminary and fast weight estimation, only the first level can be executed. In this level the structure is considered as concentrated elements, no-shear lag effects are included, the honeycomb sandwich skin is simulated by an equivalent solid laminate structure taking in account only its in-plane and bending behavior, neglecting in this way the transverse shear effects. For a more detailed design (core thicknesses and layups) the second and third level are necessary: in the second one, the core effect is considered; in the third one, the finite element analysis is executed and an iterative approach FEA/structural optimization is applied.

The MLISS procedure guarantees the wing structural sizing in a very fast way; this aspect is very important when it’s necessary to iterate the structural sizing many times as it occurs in an aeroelastic wing design. The procedure gives
accurate results and reduces the computational costs compared with the standard optimizations using only general-purpose software.

The future aim is to extend this procedure to other wing concepts as for example stiffened skin panels, and to other structural items too, as fuselage and vertical fin.

References