Structural design of a composite aileron using a multi-step integrated procedure

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Abstract

This work describes the structural design of a composite material aileron of a business aircraft with the target of weight reduction with respect to the metallic reference baseline. It proposes a multi-step procedure for the design and analysis of a composite material structure.

A carbon-epoxy material is used for the structural item. An integrated procedure (FEM/analytical and computational formulations) for the design and analysis is developed. In the first level the structural item is considered as concentrated elements. The internal loads are evaluated by elementary theory and a preliminary layup configuration for the structural components (skin, spar, etc.) is chosen by means of a stand-alone approach using a structural sizing software. In the next step a finite element model of the structural item is developed with the preliminary layups, and a general-purpose finite element software is used to evaluate the internal FEA loads acting on the different structural components. Finally the finite element model (geometry and internal loads) is imported into the structural sizing software, which chooses, for the different structural parts, the best layup satisfying the minimum weight requirement. The iterative procedure *FEM/structural sizing software* is defined and it runs until a convergent solution is obtained. The aileron is designed at ultimate loads and a weight reduction of about 14% respect to the metallic baseline is achieved. The skin and the spar are made of solid laminate and a foam material is used at the trailing edge for shape stability according to RTM technology constraints.

Keywords: aileron, solid laminate, multi-step procedure, margin of safety, weight reduction, composite material structure.

1 Introduction

The background of the aerial passengers and commercial transport is undergoing a deep evolution in all its aspect: market, logistics and organization, technologies. In particular concerning the aviation market of business the management of fleets of little aircraft in the form of Aerotaxi and of Fractional Ownership is in fact a reality in strong expansion in the USA and it is also affirming on the European market. This new customer typology necessarily is paying much more attention to the costs of purchase and of management. The commercial traffic also is undergoing a strong evolution due to the development of the activities of the carriers and of the delivery of bought on-line goods. The volume of the single shipment decreases quickly because of the growing use of the express carriers and therefore the traffic of the aerial shipments will be trebled within 2019. The request of little aerial vectors turns out growing in such sense for the commercial transport. The new builders of aircraft have understood the necessities of developing new structures according to new criteria and highly innovative technologies. In particular the research activities will have the task to conceive a constructive highly modular architecture; the effects of the modular structure of the architecture will be reflected on the production costs thanks to a reduction of the number of parts and of manufacturing and assembly time. The development and the manufacturing of structural components in *advanced composite material* besides to be often an advantage under the point of view of the reduction of the weight, costs and the times of manufacture, very often is a forced choice for the maintenance of project requirements, in terms of stiffness and deformability, or the maintenance of forms and complex curvatures which they often turn out impossible to realize with metal structures. A500 by Adam Aircraft Industries Inc., and Premier I by Raytheon Company are clear examples of how an "all composite" aircraft can be built and endorsed, after the FAA certification, for flight.

This work deals with the design, in composite material, of an aileron of a business aircraft; the reference baseline is the left wing aileron, in metallic material, of *P180 Avanti* of *Piaggio Aero Indutries*; The control surfaces generally experience considerable loads in certain critical flight conditions; for this reason this kind of structures is sufficiently complex in terms of stiffness and in terms of assembly parts cost.

The work has been developed by CIRA and Piaggio Aero Industries in the framework of VITAS project, a research program financed by the Italian Ministry of Research.

For the same dimensions and load acting on the metallic baseline, the design in composite material at ultimate load according to no-buckling and no-strength failure is developed, and also according to the constraints due to manufacturing process expected: RTM (Resin Transfer Moulding). While the metallic baseline is a full-depth honeycomb, the composite design foresees a no full-depth configuration but it's a solid laminate for the skin and for the spar with foam material at the trailing edge to satisfy the shape stability. In the design an high strength carbon-epoxy material is used; a knockdown factor (about 50%) is applied to its strength properties just to take in consideration the moisture and damage impact effects.

For the composite design a multi-step integrated procedure is used; multilevel approach for structural optimisation is becoming an important target for the aerospace design (Carrera *et al* [1], Gasbarri *et al* [2], Liu *et al* [3]). Fixed the structural configuration, from the external loads are determined the internal loads acting on the different structural parts of the aileron (spar, skin, etc) by elementary theory; the structure in this case is as concentrated elements and with a stand-alone approach, using the structural sizing software *Hypersizer* (Collier Research ver. 3.7.1), a preliminary layup is individuated. With this preliminary layups a FEA model of the aileron is build and the internal FEA loads acting on the different structural components are evaluated by using MSC/NASTRAN software. The FEA model, geometry and internal loads, is imported in Hypersizer, which chooses, for the different structural parts, the best layup satisfying the minimum weight requirement. For these new layups the FEA internal loads are changed and then a closed loop *Nastran/Hypersizer* is applied until a the convergent solution is obtained.

2 Metallic reference baseline

The actual aileron (fig.1) is a metallic sandwich, with metallic honeycomb of the type CR ¼-1-5052-U-60 covered by sheet of aluminium alloy type 2024-T3 of thickness 0.4 *mm*. The honeycomb section varies between a maximum thickness of 54 *mm* up to a minimum thickness of 7 *mm*. The front spar and the closing ribs are made in aluminium alloy 2024 T42 with a thickness of 0.6 *mm*. The aileron dimension are:
max length 1700 *mm,* max width 190 *mm*.

In fig.2 the spar section view is shown. Table 1 shows resultant load acting on the aileron. The weight of the structure is 2.5 Kg.

Table 1: Critical load on aileron.

Ultimate design load factor	Design load condition	Ultimate design load
3.25	Maximum up load roll manoeuvre at VC, W=5240 Kg	Fz = 6103 N

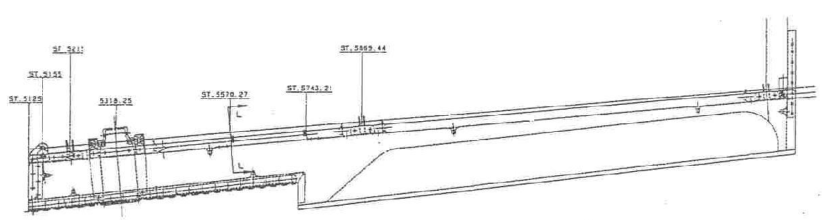

Figure 1: Metallic aileron drawing.

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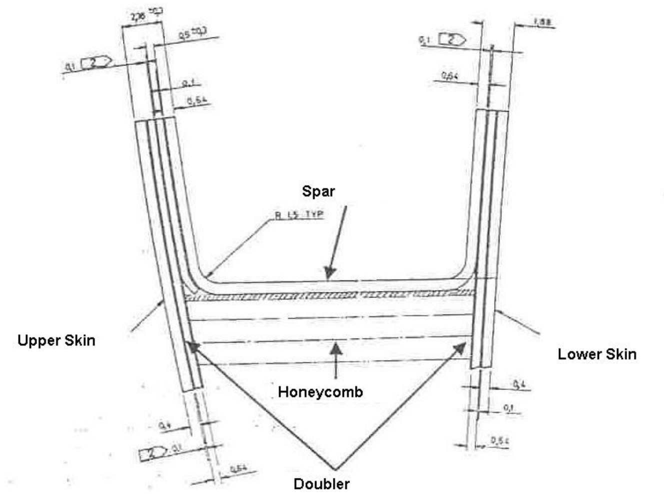

Figure 2: Metallic aileron section.

3 Material and methods

The composite aileron design is developed at ultimate load according to
a) no-buckling and no-strength failure;
b) weight reduction respect to the metallic baseline;
c) rotation at tip less than 3.5 degrees;
d) reduction of number of parts.

The aileron is devised in solid laminate; fig. 3 shows the aileron configuration selected that reduces at minimum the difficulties and the problems of the RTM process. There are only two structural parts bonded by means of structural adhesive: PART 1, as box composed of the upper and lower skin, ribs and trailing edge; PART 2, spar and tapered leading edge. In terms of design requirements, this aileron configuration implies the same layup for the upper and lower skin and the same layup for the spar and the leading edge.

3.1 Methods

An integrated and multi-step procedure for the design is developed.

In the first step, the aileron in metallic material is considered (reference baseline). From the ultimate external loads acting on the aileron, the internal loads, shear flows and normal stresses, are evaluated by elementary theory. The structural item is considered as concentrated elements: the skin and the web of the spar withstand only the shear flows, while the caps the normal stress due to bending.

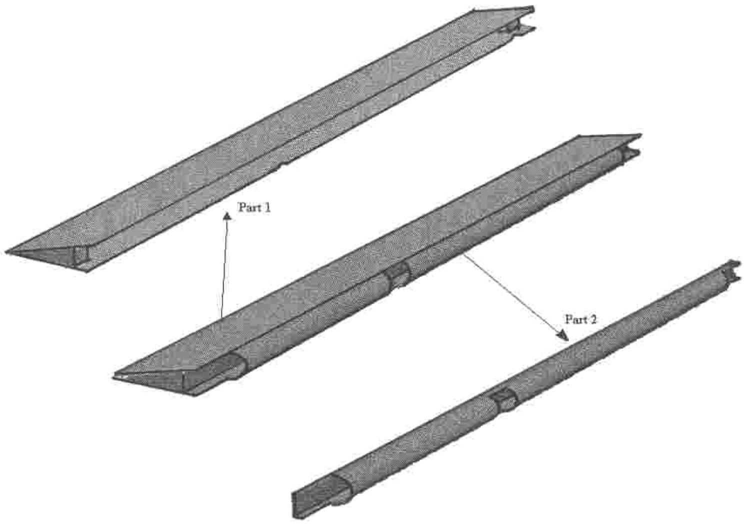

Figure 3: Aileron configuration.

At this point the internal loads evaluated, acting on the skin and the spar in metallic material, are used to size the same components in composite material using semi-empirical formulas and principally the structural sizing software Hypersizer means of a Free Body Diagram approach [4]. In this approach Hypersizer uses manual input of loads.

Based on the panel length, width, concept, shape, material and layups, this software 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 of the panel segments are integrated and summed to verify equilibrium of forces and strain compatibility for the panel/beam as a whole. The panel/beam designs are evaluated for strength and stability for the applied loads and boundary conditions using analysis methods based on traditional industry methods, modern analytical and computational solutions.

In this case the aileron structural components are considered in solid laminate and, assigned the composite material properties, a layup optimisation for these components is achieved considering a large number of candidate layup families (also than more 400 layups): the software's optimiser has determined for the spar and the skin the lightest layup with all strength and stability margin of safety ($MS = \frac{P_{critical}}{P_{actual}} - 1$) positive.

In the second step the layups determined previously are used as a first attempt in the aileron composite FEA model, and the MSC/Nastran finite element

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software is used to evaluate the internal FEA loads acting on the different structural components of the structure [5].

In the third step the finite element model (geometry and internal loads) is imported in Hypersizer, which uses FEA computed internal loads [6]. This permits the automation of the full design, analysis and build process. The FEM is used to resolve the general boundary conditions and the applied external loads into internal loads that are used to size the panels and/or beams. Then the structural sizing software creates generalized elastic stiffness terms to send back to the FEM for another iteration of computed internal loads path, and are analysed and optimised structural components, which are pieces of panels and beams. Many finite elements are used to model a structural component, so designing assuming the maximum element load could be far too conservative and results in overweight design; for this reason statistical methods are used to determine the appropriate design-to loads: a standard deviation factor for determining the design-to loads for strength analysis can be selected. The instability (buckling) instead depends on an integrated and compressive type load rather than an element peak load, so a different statistical approach is used: the percentage of the component's area that is in the compressive buckling zone is statistically determined and the compressive magnitude is integrated over that area.

The third step is the starting point of an iterative procedure FEA/Hypersizer that runs until a convergent solution is obtained (according to design requirements): there are three times for

- reading FEA internal loads,
- performing a minimal optimisation,

and

- updating the FEM.

Regarding the FEM, the external pressure load is applied with a triangular distribution along the chord (with the resultant applied at 1/3 of the chord) with decreasing values from the hinge line to the trailing edge, so that the resultant load is that one of table 1. In the aileron actuator section a torsional fitting is applied in order to simulate the real behaviour of the structure.

For the final optimised FEM by means of a modal analysis the aileron stiffness is verified; the existence of the rotational rigid mode around the hinge line and the first elastic torsional mode are evaluated: the rotation at aileron tip less than 3.5 degrees and a torsional frequency of 102 Hz are indexes of a good torsional stiffness of the aileron. Finally a beam equivalent model is developed to verify analytically these results.

3.2 Materials

The aileron material is an high-strength carbon-epoxy material: fibre *HTA 5131,* matrix *RTM6*; the property values and the ply thickness (obtained by means of internal laboratory tests on RTM unidirectional laminate panels) are shown in table 2; a knockdown factor is applied to the strength properties to take in consideration moisture and impact damage effects.

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Table 2: Aileron carbon-epoxy material properties.

density	1435.5	*Kg/mm^3*
V_f	55%	
V_m	42%	
V_{void}	3%	
E_2^t	9.16	*GPa*
E_2^c	9.16	*GPa*
F_2^{tu}	86	*MPa*
F_2^{cu}	191.8	*MPa*
ε_2^{tu}	0.0094	
ε_2^{cu}	0.0209	
E_1^t	126	*GPa*
E_1^c	126	*GPa*
F_1^{tu}	1692.8	*MPa*
F_1^{cu}	1246.4	*Mpa*
ε_1^{tu}	0.0036	
ε_1^{cu}	0.0036	
G_{12}^s	4.88	*GPa*
F^{su}	104.96	*MPa*
γ_{12}^u	0.0215	
ν_{12}	0.34	
Ply thickness	0.30	*mm*

4 Results and discussion

The final aileron structural configuration is characterised by only two ribs, one is applied in the actuator section (at aileron root) and the other one in aileron tip: buckling analysis have verified that no other ribs are necessary to assure no-instability failure.

The layup configuration, optimised by the design approach discussed in § 3.1, is shown in table 3 and in table 4 the corresponding minimum margins of safety, for the spar and the skin, are indicated: as shown, the aileron controlling failure mode is the panel buckling. About the aileron trailing edge, the design requirement of a total skin thickness of 3.2 *mm* (upper + lower skin) has made it necessary to reduce the plies of the skin to only 2 plies on the trailing edge, for both the upper and the lower skin, according to no-strength failure requirement. In fig. 4 is summarised the skin layup configuration according to the requirement discussed above, and in table 5 is shown the maximum *Tsai-Hill* failure index of the plies on the trailing edge (Di Palma *et al* [7]); moreover, to assure the trailing

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edge shape stability, a foam material is used as filler (table 6): Roachell 110WF (Niu [8]).

Table 3: Layup configuration.

	layup	laminate thickness
		(mm)
skin	5_[90/-45/0/45/90]	1.5
spar & leading edge	4_ [0/45/0/90]	1.2
ribs	4_ [0/45/0/90]	1.2

Table 4: Minimum margins of safety.

MS	structural component	failure mode
0.06344	lower skin	Panel buckling
0.1301	spar	Panel buckling

Table 5: Tsai-Hill failure index on the trailing edge.

Layup	σ_1	σ_2	σ_{12}	Tsai-Hill
	(N/mm^2)	(N/mm^2)	(N/mm^2)	
45/90	55	2.2	1.12	0.01

Table 6: Roachell 110WF.

density	110.72	*Kg/mm^3*
E	0.17	*Gpa*
G	0.0567	*Gpa*
F^{tu}	3.62	*Mpa*
F^{cu}	3.52	*Mpa*
F^{su}	2.35	*Mpa*

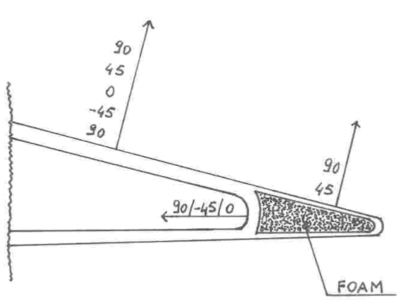

Figure 4: Skin layup configuration.

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The resultant weight structure, carbon-epoxy plus foam material contribution, is 1.9 *Kg*; it's 76% of the metallic weight structure. This value doesn't include the metallic mesh-screen contribution and the local application of fibreglass plies: for the composite structures the use of a metallic mesh-screen is necessary to assure the lightening protection, while fibreglass plies are used to avoid galvanic effects where carbon and metallic materials are in contact, as the joint areas. For these reasons a mesh screen with a density of 2.17E-6 Kg/mm^3 and a thickness of 0.10 *mm* is considered, while for the galvanic effect a fibreglass ply of 0.14 *mm* of thickness with a ply density of 1.44E-6 Kg/mm^3. The additional weight structure is resulted of 0.26 *Kg*, that is a total aileron weight structure of 2.16 *Kg* with a weight reduction of about 14% respect to the metallic baseline.

5 Conclusions

The paper has proposed the design, in composite material, of an aileron of a business aircraft by means of a multi-step and iterative procedure aimed to the optimisation of the layup configuration. The results obtained considering the structure as concentrated elements (step 1) are lightly different from those obtained by a more detailed analysis importing the FEM in the structural sizing software (step 3); the more detailed level that characterizes the second approach occurs, so as significant, only in the different ply orientation of the skin and the spar: in the first approach are not included the *shear-lag* effects. The procedure described in § 3.1 gives accurate results and reduces the computational costs compared with the standard optimisations using only a general purpose software. The aileron is designed at ultimate loads taking in consideration the moisture and impact damage material effects; the sizing criteria is resulted the panel buckling: on this criteria, strength material properties have a little effect, are important the elastic properties and the stacking sequence of the plies. Respect to the actual metallic baseline is reached a weight reduction of about 14% with only two parts to assembly; this last aspect is very important in terms of assembly/parts cost reduction respect to the metallic baseline: even if RTM tooling and facilitation costs can be higher, the reduction in recurring costs can be considerable; a build rate of one day usually maintains the manufacturing cost at an optimum level.

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