DESIGN AND ANALYSIS OF A COMPOSITE FUSELAGE

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Abstract *The present study presents a new methodology developed for an analytical model of a composite fuselage. It also presents finite element analyses of a simplified model and comparisons with more complete models. These comparisons show that there is a very good correlation between both models for the cases studied. Therefore, the applicabilty of the proposed procedure was demonstrated. Based on finite element analyses, the present paper also presents a weight comparison between a composite fuselage and an aluminium alloy one.This comparison assess the weight reduction obtained with the use of composite materials for designing the fuselage.*

Keywords: composites, fuselage, design procedure, finite element

1. INTRODUCTION

Aircraft manufacturers have been gradually increasing its reliance on composite materials. For example, Boeing 777 featured an all-composite empennage and composite floor beams. Nevertheless, the composite materials community is very much aware of the cost implications of introducing more composite materials. It was only when a technological breakthrough on the manufacturing side came about that it considered widespread use of such materials, for example, in the Boeing 787. Basically, this involves the same fiber-resin system as used in the Boeing 777 empennage but with radically different automated fiber-placement techniques. These techniques allow rapid and accurate positioning of fibers onto a mandrel that will initially create the stringers and then apply the fuselage skin to varying thicknesses, as desired. Each fuselage section is then autoclave cured and the mandrels are then disassembled and removed. The Boeing 787 fuselage is built in five main sections and composite materials that account for 50% of the aircraft's total structural weight. (Aircraft Technology Engineering & Maintenance, 2005)

Both Boeing and Airbus have recognized that they have the opportunity to increase the thickness of composite structures where there is a high probability of impact damage. Areas such as doors, door surrounds, wing tips, wing leading and trailing edges and wing-to-body fairings are all prone to ground vehicle impact damage and increasing the thickness of any composite structures in these areas should reduce the probability of significant damage. The possibility of replacing damaged components at these locations still remains where the designs permit. (Aircraft Technology Engineering & Maintenance, 2005)

Boeing intends to capitalize on in its 787 CFRP fuselage design as that it can work with larger pressure (from a cabin altitude of 8,000ft to a cabin altitude of 6,000ft) without adding substantial weight to the airframe structure. Furthermore, in view of the excellent corrosion resistance of advanced composites, Boeing is also contemplating the introduction of a cabin humidifier, also intended to make the flight experience a more pleasurable one. Finally, Boeing intends to make the windows on the 787 significantly larger than traditional windows. Airbus has claimed that it intends to do the same in each of these areas on its A350 (Aircraft Technology Engineering & Maintenance, 2005; Wall, 2005).

The one-piece, business jet fuselage, designed by Dassault Aviation in conjunction with BAE Systems, was manufactured using pre-impregnated carbon fiber slit tape and honeycomb core. Automated fiber placement enables manufacturability of a single-piece fuselage that can replace typical business jet structures made up of many individual components and thousands of fasteners (Leininger, 2005).

The main scope of this paper is to present an analytical method of buckling analysis of laminated composite fuselage. This method was developed based on different references and it is demonstrated by the comparison between the analytical method results and the finite element analysis method results. A comparison between an aluminum fuselage and a composite fuselage is also presented showing the less weight advantage of the composite fuselage.

This paper presents, as an example, the solutions of the structures design of a fuselage for a 30 seats commercial aircraft. Among the solutions are the interior layout and cross section definition, initial geometric and weight sizing and aircraft configurations. All the solutions were studied considering the aerodynamic area and the structures area and the integration of all aircraft systems.

The aircraft main characteristics are: an all-composite semi-monocoque type fuselage, with metallic low-wing, conventionally tailed, pressurized aircraft featuring a retractable twin wheeled, tricycle-type landing gear system and two high bypass ratio under the wing mounted turbofan engines. Figure 1 shows a sketch the aircraft.

Reparability and interchangeability of parts to lower production and maintenance costs were a vital part of the design process. Wing primary structure is manufactured with aluminum alloys, with formed and machined parts. Fuselage primary structure is Carbon Fiber Reinforced Plastic (CFRP) with Fiber Placement (FP), Resin Transfer Molding (RTM) and pultrusion manufactured parts. Control Surfaces are also manufactured in composites (CFRP) to reduce weight and optimize manufacture.

Figure 1. Aircraft's sketch

2. FUSELAGE

The proposed aircraft fuselage structure is a innovative fuselage concept. The whole fuselage is fabricated with Carbon Fiber Reinforced Plastic (CFRP). The main advantages in this new design are: (1) very good integration; (2) faster fabrication and assembly; (3) weight reduction (10-15%); (4) possibility of thickness variations; (5) less waste of raw material; (6) higher passenger comfort level; (7) possibility of larger windows; (8) longer structural life (less sensitive to fatigue).

There are also some disadvantages, although there are some possible solutions to overcome these disadvantages. The main disadvantages are: (1) electro-magnetic interference); (2) return of electrical current; (3) lightning protection; (4) higher machinery investments; (4) higher certification costs.

The fuselage will be constructed in three parts along the longitudinal axis in order to facilitate the construction process and improve reparability. Each part of the fuselage will be manufactured by the FP (Fiber Placement) process resulting in a single non-circular panel. All the stringers will be positioned in the mandrel of the ATL process and these stringers will be already fabricated and cured at this process stage. The result of the FP process will be the stringers mounted in the single non-circular panel skin. The next fabrication process is the panel skin cure.

It is important to emphasize that the utilization of composite materials gives a possibility off thickness variation in almost all parts where they are used. This is extremely useful and is widely used in the present design, particularly where the aircraft presents the highest loads and stresses.

The fuselage frames have special shape. The frames have a "C" shape with an integrated flange in the opposite direction of the "C" opening and will be fabricated by the RTM (Resin Transfer Molding) process. The frames will be fabricated in three pieces in order to be positioned and mounted correctly inside the fuselage section. Despite these special frames having high costs, they have improved mechanical properties and will contribute for the aircraft weight reduction. In the wing-fuselage mounting section, we must take out the extra flange in order to make the mounting process feasible.

The next step will be the mounting of the floor structure. This process starts by the fabrication of a structure where the cross beams and the vertical struts are integrated and are fabricated in CFRP by RTM process. The next step will be the mounting of the longitudinal floor beams, which are already fabricated and cured. The pultrusion fabrication process will be used because the constant section of the longitudinal floor beams. Next to the floor beams mounting process is the seat tracks beams mounting process. This will be the only kind of metallic material on the fuselage and the reason for the use o a metallic material is the electric current return. The metallic material chosen for this application is Al 7050. After the mounting of the floor structure, the mounted sections of the fuselage will be joined and the last steps of the fuselage mounting process will continue as usual in the aircraft industry.

The overall layout of the fuselage structure is presented in Fig. 2.

Figure 2. Fuselage structure overall layout

2. BUCKLING ANALYSES

The main components of the aircraft's fuselage were initially designed with estimated thicknesses. Hence in this section it will be presented the buckling analysis made to the fuselage. The buckling analysis was the only analysis made to the fuselage because buckling is in the most cases the most critical condition to the fuselage.

The design moment and shear load of the fuselage are $M_y = 1.03 \times 10^6$ Nm and the $S = 168050$ N, respectively. The fuselage diameter is 2280 mm and the distance between frames is 432 mm.

The buckling analysis consists of two main types of analyses. The first one is an analytical analysis based in an arrangement of references as Boeing's design manuals (1994, 1995), MIL-HDBK-17-3F (2002), and Bruhn (1973). The second one is a finite element analysis that was made in order to compare to the results of the analytical model. Niu (2002) and Roskam (1997) also present useful material for the design of fuselages.

The carbon/epoxy materials used are: (1) pre-preg tape MEP-15-047 and (2) pre-preg plain weave MEP15-047. The mechanical properties of these materials are presented in Tab. 1. Due to the several types of loads that an aircraft's fuselage skin supports such as pressure, compression, tension, shear and torsion, the stacking sequence shall be quasi-isotropic.

The stringer of the aircraft has a "T" shape in order to be simple and cheap to manufacture and is also very common in the aerospace industry. The stringer is composed of layers of plain weave carbon fabric; the stringers are reinforced in some cases by layers of pre-preg tape. The width of the stringer is 40 mm whereas its height is 20 mm.

A few considerations were made in order to simplify the analytical model. The same hypotheses were also made in the finite element analyses in order to compare the results of each type of analyses: (1) the cross-section is considered circular; (2) there are no windows; (3) the aircraft's passenger floor is not accounted for, (4) the stringers are equally spaced, and (5) the skin and stringer joining is considered perfect.

2.1. Analytical Model

The first step of the analytical model is an estimative of the thickness, number of layers and stacking sequence of the skin laminate. Then, the skin cross section area and the moment of inertia of the cross section about axis y-y can be computed. Also, the effective Young's modulus and the Poisson ratio of the skin are computed. Next, an estimative of the thickness and number of layers of the stringer is made. As width, height and number of stringers are known, the cross section area and the moment of inertia of the cross section about axis y-y of the stringer can be computed.

The following steps are used in order to calculate the local compression buckling stress F_{cc} and local compression stress f_c of the stringer. First, the following parameters are computed:

$$\kappa = \frac{bD_{11}}{tE_{xc}}\sqrt{\frac{F_{cu}}{\sqrt{E_{xc}E_{yc}}}} \qquad \text{where} \qquad \bar{E} = \frac{12\left(1-v_{xy}v_{yx}\right)D_{11}}{t^3} \tag{1}$$

With the value of κ, one can obtain the value of $\left(F_{cc} E_{xc} \right)/\left(F_{cu} \bar{E} \right)$ from a figure available in Bruhn (1973) and then compute F_{cc}. The design procedures described in Bruhn (1973) are applicable for metallic structures. However, as the laminates used in the fuselage are quasi-isotropic, these procedures are adequate for a preliminary design.

Table 1. Mechanical properties of the carbon/epoxy unidirectional tape and fabric

property	unid. tape	fabric
Thickness per ply (mm)	0.185	0.21
Density (g/cm^3)	1.79	1.57
Longitudinal modulus, E_1 (GPa)	137.3	62.6
Transverse modulus, E_2 (GPa)	7.8	59.3
In-plane shear modulus, G_{12} (GPa)	5.23	4.6
Poisson's ratio, ν_{12}	0.36	0.062
Longitudinal tensile strength, F_{1t} (MPa)	2057	621
Transverse tensile strength, F_{2t} (MPa)	46.9	594
Longitudinal compressive strength, F_{1c} (MPa)	1610	760
Transverse compressive strength, F_{2c} (MPa)	207	707
In-plane shear strength, F_6 (MPa)	135	125

The following steps are used in order to compute the local compression buckling stress F_{cc} and local compression stress f_c of the panel. The moment of inertia of the stringer cross section about axis y-y, I_{st}, is computed and then the stiffener radius of gyration ρ is computed as: $\rho = \sqrt{I_{st}/A_{st}}$. Next, the compression stress when L'/ρ is equal to zero, is determined:

$$F_{co} = \phi\left(1 - \sqrt{1 - \left(2F_{cc}\right)/\phi}\right) \quad \text{where} \quad \phi = \left(2\pi^2 E_{xc}\right)/12.5^2 \tag{2}$$

Then, the buckling stress of the panel F_c: is determined from: $F_c = F_{co}\left[1 - \left(F_{co}\left(L'/\rho\right)^2\right)/\left(4\pi^2 E_{xc}\right)\right]$

The procedure continues with the computation of the effective width of the skin, $b_e = 1.70t\sqrt{E_{xc}/f_c}$, the effective area of the skin $A_e = b_e t$ and the parameter $\xi = A_e/A_{st}$. Next, the distance from the stringer center and the surface middle surface e is computed and the radius of gyration of the stringer-skin combination, ρ, is computed from:

$$\rho = \rho_0 \sqrt{\left(1 + \xi\left[1 + \left(e/\rho_{st}\right)^2\right]\right)\Big/\left(1 + \xi\right)^2} \tag{3}$$

A new value of the buckling stress of the panel Fc is determined. An iteration process of Fc is repeated until the value of F_c converges. The compression stress f_c of the panel is computed:

$$f_c = \left(M_y z\right)\Big/\left(I_{yy(skin)} + \sum I_{yy(stringers)} + t b_e z^2\right) \tag{4}$$

With the F_c compressive buckling stress value of the panel and the panel compression stress is possible to calculate the margin of safety: $MS = F_c/f_c - 1$.

The following steps are used in order to compute the buckling of panels between stringers and frames. As the number of stringers, radius of the fuselage and distance between frames are known, the dimensions of the panel can be defined: $b = \pi D/(number\ of\ stringers)$ and $a = (distance\ between\ frames)$. With these parameters one can compute the parameter:

$$\vartheta = \frac{b^2}{rt}\sqrt{1 - \mu^2} \tag{5}$$

Where μ is the Poisson ratio of the laminate. Then, the parameters K_c and K_s can be found from figures available in Bruhn (1973). With this value the compressive and shear buckling strengths can be computed as:

$$F_{cr} = \left[\frac{K_c \pi^2 E}{12(1-\mu^2)} \right] \left(\frac{t}{b} \right)^2 \tag{6}$$

$$F_{scr} = \left[\frac{K_s \pi^2 E}{12(1-\mu^2)} \right] \left(\frac{t}{b} \right)^2 \tag{7}$$

The compression stress f_c of the panel is computed from:

$$f_c = \left(M_y z \right) / \left(I_{yy(skin)} + \sum I_{yy(stringers)} \right) \tag{8}$$

Then, the compression stress ratio is computed $R_c = f_c / F_{cr}$. Also, the shear stress superior and the shear stress inferior are computed, respectively, from:

$$S_{SS} = \frac{V_z}{I_{yy(skin)} + \sum I_{yy(stringers)}} \frac{z A_{st}}{t_{skin}}$$

$$S_{SI} = \frac{V_z}{I_{yy(skin)} + \sum I_{yy(stringers)}} \frac{z \left(A_{st} + A_{sk} \right)}{t_{skin}}$$

Find the shear stress average, $f_s = (S_{SS} + S_{SI})/2$, and compute the shear stress ratio: $R_S = f_s / F_{scr}$. With compression stress ratio R_c, shear stress ratio R_S and the panel compression stress is possible to compute the margin of safety MS:

$$MS(\%) = 100 \left(\frac{2}{R_c + \sqrt{R_c^2 + 4R_S^2}} - 1 \right)$$

2.2. Finite Element Analysis

The finite element analysis was performed following the steps describe below. The fuselage structural design was modeled using the software CATIA V5. Taking into consideration the most critical bending load and shear load, the most critical sections of the fuselage structure were designed in more detail. This design was made in order to create and export the finite element mesh.

In order to analyze the fuselage mesh, this mesh was imported using the software MSC NASTRAN. A mesh was created retaining just the essential data for the specific analysis proposed. The fuselage section mesh is presented on Fig. 3. After this, the loads and constraints were applied. The constraint applied was fixed nodes in one side of the fuselage section. The applied loads were the same as in the analytical model. This model has 8640 elements and 7601 nodes. This was the model that was used in all the finite element analyses.

The eigenvalues that results from the finite element buckling analysis that present values below 1.00 mean that the panel buckles. On the other hand, if the values are above 1.00 it means that the panel does not buckle. The first six eigenvalues were computed from the finite element model. The first eigenvalue is 1.018866, which means that the panel does not buckle. The second eigenvalue is negative equal to -1.019301. A negative value means that the structure also buckles when loaded in tension. The rest of the eigenvalues are always higher in modulus than those; this means that they should not be considered in the design.

A study was made in order to check if the used finite element model accurately predicted the buckling of the structure. First, four new finite element models were created. The first model created was a less refined model than the original. The second model was a model created with a mesh more refined than the original with a larger number of elements. Then a model with three fuselage sections was created, where each fuselage section have the same number of elements than the original fuselage section. In the middle of two consecutive fuselage sections there was considered a fuselage frame. This fuselage frame was not accounted for in the single fuselage section model. The last model created was a model with five fuselage sections. Each fuselage section had also the same number of elements than the original one and it was also considered fuselage frames for each fuselage section joint. A model comparison is presented in Fig. 4.

Two studies were made in order to compare the quality of the original model with the new models created. Each study was made with a distinct configuration. In the first study the skin had 16 layers of tape and the stringers had 6

layers: 4 layers of fabric and 2 layers of tape. The studies results are presented in Tab. 2 and 3. In the second study the skin had 17 layers of tape and the stringers had 4 layers of fabric. In both studies, the models with 3 and 5 fuselage sections included frames and these frames were tested with different numbers of layers in order to assess the influence on the model of different stiffness of the frames.
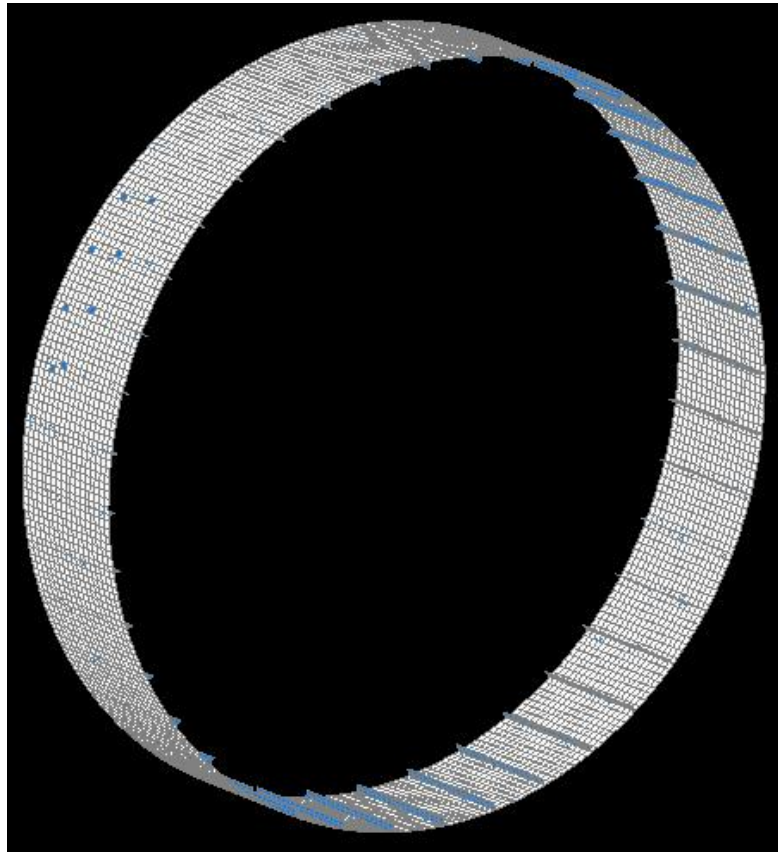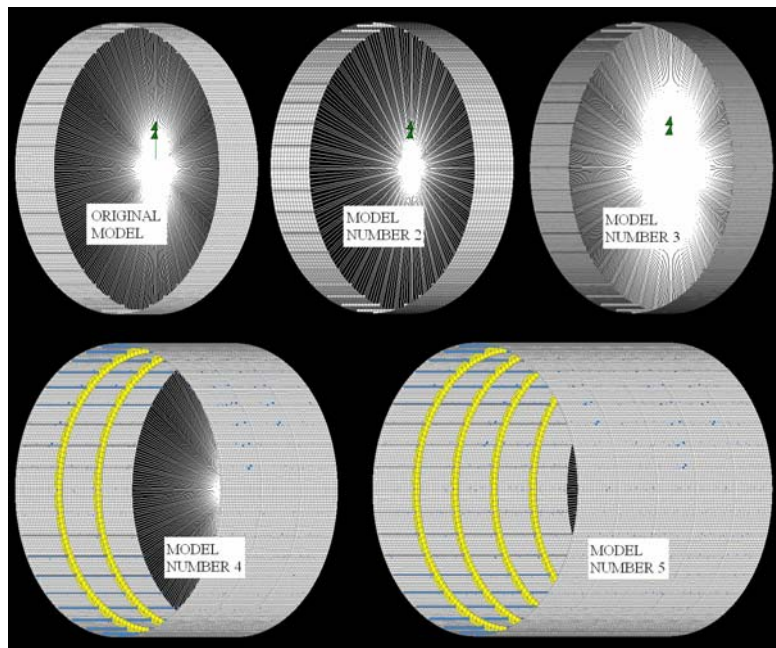

Figure 3. Fuselage section mesh

Figure 3. Models for convergence study

Table 2. Results for convergence study for 16 layers skin and 6 layers stringers

Model Number	Mesh Quality	Skin		Stringers		Frame		Fuselage	Eigen-value
		Layers	Thick. (mm)	Layers	Thick. (mm)	Layers	Thick. (mm)	Length (mm)	
1 (ref)	Regular	16	2.96	6	1.21	0	0	432	1.102
2	Poor	16	2.96	6	1.21	0	0	432	1.110
3	Refined	16	2.96	6	1.21	0	0	432	1.029
4	Regular	16	2.96	6	1.21	9	1.89	1296	1.038
4	Regular	16	2.96	6	1.21	18	3.78	1296	1.082
5	Regular	16	2.96	6	1.21	9	1.89	2160	1.030
5	Regular	16	2.96	6	1.21	18	3.78	2160	1.079

Table 3. Results for convergence study for 17 layers skin and 4 layers stringers

Model Number	Mesh Quality	Skin		Stringers		Frame		Fuselage	Eigen-value
		Layers	Thick. (mm)	Layers	Thick. (mm)	Layers	Thick. (mm)	Length (mm)	
1(ref)	Regular	17	3.15	4	0.84	0	0	432	1.166
2	Poor	17	3.15	4	0.84	0	0	432	1.181
3	Refined	17	3.15	4	0.84	0	0	432	1.128
4	Regular	17	3.15	4	0.84	9	1.89	1296	1.076
4	Regular	17	3.15	4	0.84	18	3.78	1296	1.128
5	Regular	17	3.15	4	0.84	9	1.89	2160	1.067
5	Regular	17	3.15	4	0.84	18	3.78	2160	1.122

From the comparison among the seven different tests it may be concluded that the less refined single fuselage section models (model number 2) resulted, as expected, in a larger eigenvalue than the regular mesh fuselage section (reference model) in both tests. The differences were 0.73% in the first test and 1.29% in the second. Also, the refined mesh single section fuselage models resulted in a smaller eigenvalue than the reference mesh fuselage section in both cases. The reductions were 6.6% in the first study and 3.3% in the second.

In both tests, the three sections fuselage models (model number 4) with smaller number of layers in the frame resulted in a smaller eigenvalue than the three sections fuselage with larger number of layers in the frame. This result was also smaller than the reference model. In the first test the decrease with respect to the reference model was 5.8% for the smaller number of layers in the frame and 1.8% for the larger number of layers in the frame. In the second test the decrease from the reference model were 7.7% and 3.2% for the smaller and larger number of layers in the frame, respectively. We may observe that when the frames stiffness increases, the results of the study get closer to the single fuselage section (reference model).

In both tests, the five sections fuselage model (model number 5) with the lowest number of layers in the frame resulted in a smaller eigenvalue than the five sections fuselage model with larger number of layers in the frame. This value was also smaller than the reference model. In the first test the decrement from the regular mesh fuselage section were 6.5% and 2,1% for the lowest and largest number of layers in the frame, respectively. In the second test the decrease from the reference model were 8.5% for the lowest number of layers in the frame and 3.8% for the larger number of layers in the frame. It is observed again that when the frames stiffness increases, the results of the study get closer to the single section fuselage (reference model).

From the results presented above it may be concluded that the maximum error for the simplification was about 8.5%. This means we must consider these deviations and compute the thickness adding an adequate safety margin to account for theses deviations.

3. RESULTS AND DISCUSSIONS

By comparing the analytical model results to the finite elements analyses results (keeping the loads fixed and varying the number and orientation of skin layers and stringer layers) it can be obtained the best skin × stringer relation in order to achieve the lightest fuselage section structure. These combinations analyzed are presented in Tab. 4 and 5. Table 4 presents the results of the analytical model and Tab. 5 presents the results of the finite element analyses.

The analytical model results show that it is not possible to have combinations for fewer than 14 and more than 17 skin layers because for less than 14 layers panel buckling occurs and for more than 17 layers occurs local buckling. The same situations happen for the finite element analyses results, though the analyzed numbers of layers are different. The

limits are 11 and 18 layers, respectively. The reasons that explain these limits are the same as for the analytical model analyses.

Figure 4 shows the relation between the number of layers of the skin versus the number of layers of the stringer considering the finite element analyses and the analytical analyses. The figure shows that there is a good correlation between the finite element analyses and the analytical model when the number of skin layers is as large as possible. It also shows that when the number of skin layers decreases the correlation between finite element analyses and the analytical analyses gets worse and can not be considered acceptable.

It is important to notice that the correlation between the analytical method is better when the fuselage sections are near the lightest weight. This is important because it shows that the analytical method presents good results or good correlations in the situations desired: the least weight fuselage sections.

Table 4. Analytical model results

Skin		Stringer		Fuselage section	
Layers	Thickness (mm)	Layers	Thickness (mm)	Length (mm)	Weight (kg)
14	2.59	22	4.47	432	21.92
15	2.78	11	2.21	432	19.60
16	2.96	7	1.42	432	19.34
17	3.15	5	1.00	432	19.71

Table 5. Finite element analyses results

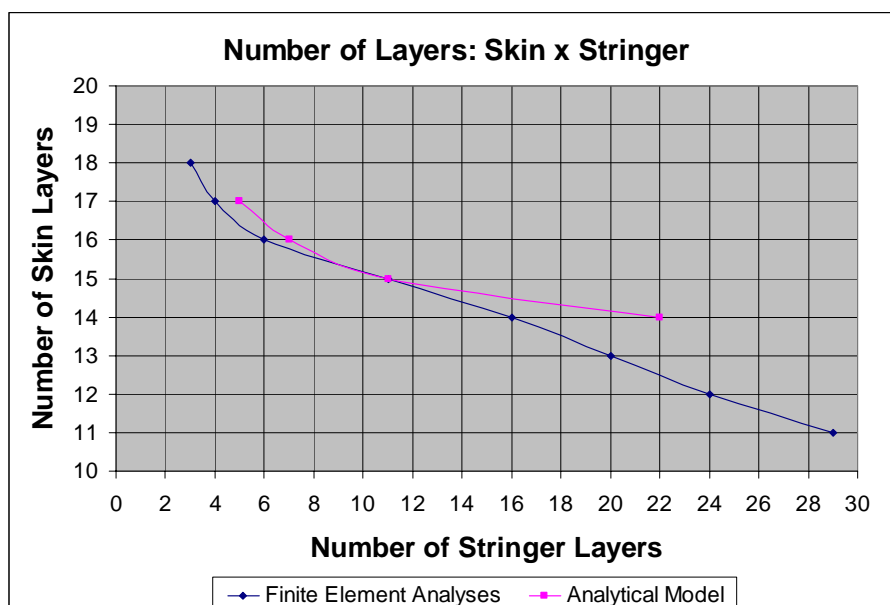| Skin		Stringer		Fuselage section	
Layers	Thickness (mm)	Layers	Thickness (mm)	Length (mm)	Weight (kg)
11	2.04	29	5.94	432	20.81
12	2.22	24	4.89	432	20.44
13	2.41	20	4.10	432	20.30
14	2.59	16	3.26	432	20.13
15	2.78	11	2.21	432	19.60
16	2.96	6	1.21	432	19.01
17	3.15	4	0.84	432	19.37
18	3.33	3	0.63	432	20.06

Figure 4. Analytical and numerical results as a function of the number of layers in the skin and stringer

4. COMPARISON BETWEEN COMPOSITE FUSELAGE AND ALUMINUM FUSELAGE

It is important to compare the design weights of the composite fuselage section and the aluminum fuselage section. The stress analyses were made by the finite element method. As the results of the composite fuselage section were presented in only the aluminum fuselage section is analyzed. The results of the aluminum fuselage section are presented in Tab. 6. The thickness of each structure is presented on Fig. 5.

From Fig. 5 it is observed that the aluminum structure is thicker than the composite structure. The weight data from the analyses are presented on Tab. 7. From Tab. 7 we may observe that the composite structures from the analytical analyses and from the finite element analyses are lighter than the aluminum structures from the finite element analyses.

Table 6. Results for aluminum fuselage

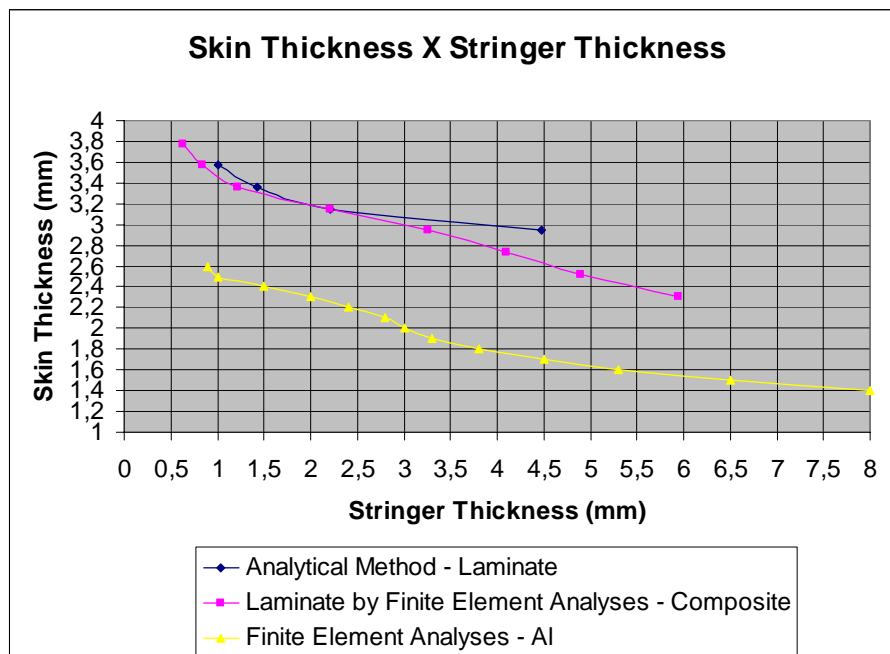| Skin	Stringer	Fuselage section	
Thickness (mm)	Thickness (mm)	Length (mm)	Weight (kg)
1.4	8	432	31.11
1.5	6.5	432	28.76
1.6	5.3	432	26.89
1.7	4.5	432	25.85
1.8	3.8	432	25.00
1.9	3.3	432	24.60
2.0	3	432	24.69
2.1	2.8	432	25.02
2.2	2.4	432	24.83
2.3	2	432	24.63
2.4	1.5	432	24.15
2.5	1.0	432	23.64
2.6	0.9	432	24.20

Figure 5. Analytical and numerical results as a function of the skin and stringer thicknesses

The results show an excessive number of layers and consequently an excessive fuselage weight. This can be improved if some different approaches are used. One of the possible solutions for the fuselage section weight reduction is to remove the consideration of equally spaced stringers. Concentrating the stringers in the most critical areas, the panel width in these areas is reduced and consequently the stress is reduced. In the less critical areas, the stress is increased. The target is to achieve equal stresses in all the areas. This would result in a fuselage with the same number of layers in all areas in the same fuselage section. This approach reduces the number of skin and stringer layers.

Another solution in order to reduce the fuselage weight is to compute the thickness required for every fuselage section and for every panel between two stringers. This solution is possible considering the FP fabrication process. The stress is reduced increasing the thickness in the most critical areas. The stress increases decreasing the thickness in the less critical areas. Again the target is to achieve equal stresses in all the areas.

Also, local buckling could be accepted, for example, above 80% of the limit load. Of course, this solution would have to be analyzed and tested in order to demonstrate that the fuselage postbuckling resistance is adequate.

Table 7. Comparison Composite Fuselage Section × Aluminum Fuselage Section

Composite Analytical		Composite Finite element		Aluminum Finite element	
Length (mm)	Weight (kg)	Length (mm)	Weight (kg)	Length (mm)	Weight (kg)
432	21.92	432	20.81	432	31.11
432	19.60	432	20.44	432	28.76
432	19.34	432	20.30	432	26.89
432	19.71	432	20.13	432	25.85
		432	19.60	432	25.00
		432	19.01	432	24.60
		432	19.37	432	24.69
		432	20.06	432	25.02
				432	24.83
				432	24.63
				432	24.15
				432	23.64
				432	24.20

5. CONCLUSIONS

From the studies presented it is concluded that the results from the analytical model and from the finite element analyses were very close and the analytical method was considered valid for preliminary design. For the more complex designs it is recommended the use of the stress analyses software because complex geometries may cause larger errors in the analytical produce.

As the fuel consumption is directly affected by the aircraft weight, the weight reduction presented from the comparison between composite fuselage section and aluminum fuselage section is the main motivation for all the studies and development of this type of material. The choice of the material should consider other factors such as: the investment on machinery, in service durability, corrosion resistance, manufacture, reparability and possibility to increase pressurization.

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7. RESPONSIBILITY NOTICE

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