Detailed design of a lattice composite fuselage structure by a mixed optimization method

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Abstract

In this paper, a procedure for designing a lattice fuselage barrel has been developed and it comprises three stages: first, topology optimization of an aircraft fuselage barrel has been performed with respect to weight and structural performance to obtain the conceptual design. The interpretation of the optimal result is given to demonstrate the development of this new lattice airframe concept for the fuselage barrel. Subsequently, parametric optimization of the lattice aircraft fuselage barrel has been carried out using Genetic Algorithms on metamodels generated with Genetic Programming from a 101-point optimal Latin hypercube design of experiments. The optimal design has been achieved in terms of weight savings subject to stability, global stiffness and strain requirements and then was verified by the fine mesh finite element simulation of the lattice fuselage barrel. Finally, a practical design of the composite skin complying with the aircraft industry lay-up rules has been presented. It is concluded that the mixed optimization method, combining topology optimization with the global metamodel-based approach, has allowed to solve the problem with sufficient accuracy as well as provided the designers with a wealth of information on the structural behaviour of the novel anisogrid composite fuselage design.

Keywords: Aircraft fuselage design; Topology optimization; Lattice structure; Metamodel; Parametric optimization.

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1. Introduction

Environmental and economic issues force future aircraft designs, like the A350, to maximize efficiencies in terms of weight and cost to keep air transport competitive and safe. As metal designs have reached their climax after 90 years of development, extraordinary weight and cost savings are unlikely to be achieved in the future (Shanygin et al. 2012). Carbon composites have high values of specific strength and rigidity, but the way in which they are currently used only provide a weight savings of 10–20% (Vasiliev and Razin 2006). Carbon fibre reinforced plastics (CFRP) have a specific strength 5 to 6 times higher and a specific rigidity 2-3 times higher than aluminium alloys (Daniel and Ishai 2005). These materials, however, demand a sophisticated layout, design, and manufacturing in order that their immense potentials could be completely exploited to demonstrate the significant advantages to metal designs for conventional aircraft airframe layouts (Ostrower 2011).

The composite lattice structure was developed by the Russian Central Research Institute for Special Machinery (CRISM) for rocket structures (Herbeck et al. 2003; Vasiliev et al. 2001, 2012; Wilmes et al. 2002) in the 1980s. These structures consist of ribs either helically or ring shaped made of unidirectional composite fibres using automatic filament winding, also known as lattice or anisogrid structures. The high mechanical properties of the unidirectional composites of the lattice ribs are the main factor for their high weight efficiency. The skin of the cylindrical or conical shells is usually manufactured from carbon fabric and only carries an insignificant part of the loading (tension, compression and shear). The automatic filament winding process ensures an integral structure with a low manufacturing cost. These experiences open up new opportunities for the optimization of composite aircraft fuselage barrels (Shanygin et al. 2012).

To determine this potential, a demonstrator was built as the DLR black fuselage concept

(Herbeck et al. 2003; Wilmes et al. 2002). However, the barrel section was never implemented in a flying aircraft. In order to address some of the issues associated with the implementation of composite lattice structure into commercial aircraft, a mixed optimization method, combining the use of topology optimization and metamodel-based approach for the design of composite lattice aircraft fuselage barrels, is utilized in this paper to transfer the concept of the geodesic composite lattice structures used in the rocket industry to the design of composite aircraft fuselages. The design of fuselage barrel sections has been achieved by a three-stage process. First, an efficient material layout (the lattice pattern of the reinforcement) of the composite aircraft fuselage barrel is identified and detailed in Section 2 by topology optimization. This technique has been widely used in aircraft and aerospace structure design (Zhu et al. 2015). Subsequently, the parametric optimization of the composite lattice structure to obtain the optimal solution describing the lattice element geometry is performed in Section 3. In this multi-parameter optimization of a lattice composite fuselage structure, one of the design variables, the number of helical ribs, is integer. It is assumed that performing a response function evaluation only for points that have discrete values of the design variables is allowed (Balabanov and Venter 2004). This makes it impossible to initially ignore the discrete nature of the design variables, solve a continuous problem and adjust the result to the given set of the discrete values, as sometimes suggested by Stolpe (2011). A discrete form of genetic algorithm (Michalewicz 1992; Bates et al. 2004) is used to search for the optimal solution in terms of weight savings subject to stability, global stiffness and strain requirements. The optimal solution has been examined by the result from the fine mesh Finite Element (FE) simulation of the lattice fuselage barrel. Based on the optimal design of the lattice fuselage barrel, the skin has finally been interpreted as a practical composite laminate which complies with the aircraft industry lay-up rules and manufacturing requirements.

Summarily, the design process of the composite lattice structure includes three phases: i) the conceptual design of an aircraft fuselage barrel is obtained by topology optimization with respect to weight and structural performance; ii) a detailed design process of an aircraft fuselage barrel is a multi-parameter optimisation problem, in which a metamodel-based optimization technique is used to obtain the optimal solution describing the lattice element geometry; iii) practical designs of the laminated composite skin complying with the aircraft industry lay-up rules and manufacturing requirements are identified. This three-phase design process will be introduced in great detail by later sections in the paper.

2. Topology Optimization

Topology optimization is a mathematical approach that optimises material layout or distribution subject to some constraints in a given design space. Its methods, theory and various applications have been recently discussed by Deaton and Grandhi (2014). In this paper, the Solid Isotropic Material with Penalisation (SIMP) (Bendsøe and Sigmund 2003) topology optimization method was used to determine the efficient fuselage stiffener arrangements. The objective of this optimization problem was to minimize the compliance of the fuselage section subject to the applied loads. Topology optimization of the fuselage structure was descripted in details by Niemann (2013). In order to clearly demonstrate the mixed optimization technique proposed in this paper, a brief introduction to topology optimization of the fuselage design has been given. The optimization problem is given by Eq. (1) and the buckling requirement was not considered at this stage.

$$\begin{array}{ll} \min & C_{Tot} \\ \text{subject to} & \sum_{n=1}^{N} \rho_n V_n \le V_0 \\ & n = 1, \ldots, N \\ & \rho_{\min} \le \rho \le 1 \end{array} \qquad (1)$$

where: C_{Tot} is the total compliance for the structure; N is the total number of finite elements in the designable part of the structure; n is the number of the analysed finite element; ρ is the design variable and artificial element density used by the SIMP method to optimize each finite element in the structure. Since the fuselage skin thickness could vary from 30 mm (full stiffener present) down to 0.1 mm (only outer skin membrane present), the value of ρ_{min} used was 0.00333. V_n is the volume of the analysed finite element and V_0 is the maximum volume of the designable structure.

The length of the fuselage section selected was 13,652 mm and it included a load and two support introduction bays, both 399.8 mm in length. The rest of the fuselage had 23 bays with a 22" (558.8 mm) pitch in Fig. 1. The cross section was made from three different radii and included a passenger and a cargo floors with struts connecting the two in Fig. 2. The loads applied were detailed by Niemann et al. (2013).

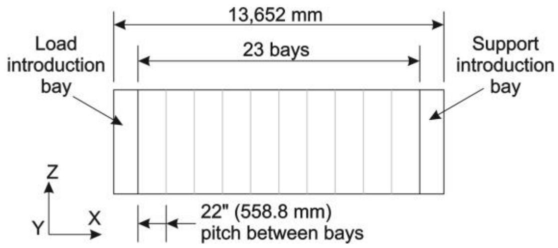


Fig. 1 Side view of the fuselage showing the different bays

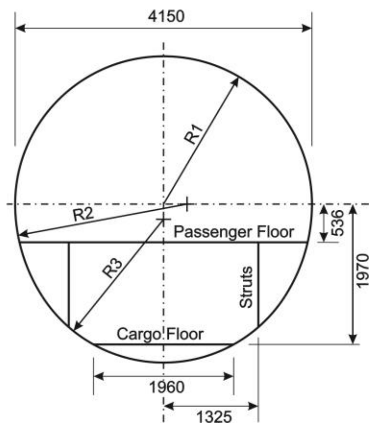
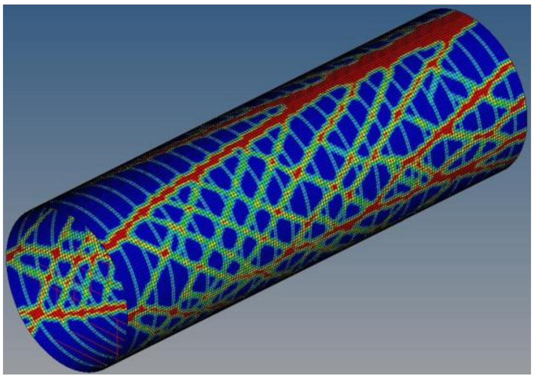


Fig. 2 Dimensioned cross-section of the fuselage section

Using OptiStruct (2013), the optimal results of the fuselage section are given in Fig. 3 and show two very distinctive features: 1) substantially sized backbones on the upper and lower extremities of the fuselage cross-section, and 2) smaller rib-like stiffeners which start and end on the backbones and also criss-cross each other at different angles ranging from 0 °to 60 °. The ideal stiffener arrangement would have continually varying angles. These would start parallel to the fuselage axis (this will function as backbones) and vary to have a maximum angle at the two sides of skin. However, these stiffeners are very complex and expensive to manufacture due to their continual varying sizes. Whereas stiffeners with constant cross-section and curvature are much easier and less expensive to produce. Taking these into account, it is reasonable to simplify these stiffeners with constant angle accordingly, then align them along geodesic lines on the fuselage skin shown in Fig. 4. This lattice structure generates non-rectangular skin bay segments in a grid-like arrangement which considerably improves the buckling resistance of the skin bay by Tan et al. (1983). The detailed research on topology optimization of the aircraft fuselage section and selection of the grid stiffener configurations for the primary aircraft fuselage design is given in the paper (Niemann et al. 2013).



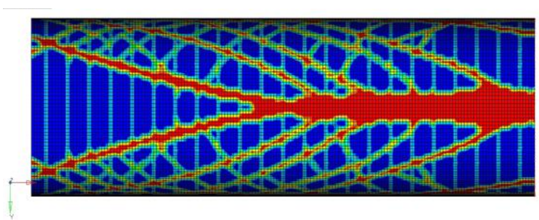


Fig. 3 Pattern of optimal reinforcement on the skin, Iso view (left) and Bottom view (right)

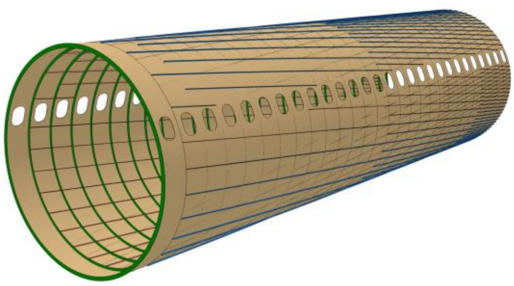


Fig. 4 Fuselage barrel concept incorporating a grid structure and using an inner skin between the CFRP-metal hybrid stiffeners

3. Parametric Optimization of the Aircraft Fuselage Barrel

Based on the results obtained by topology optimization in Section 2, the efficient load-carrying path for the fuselage structure has been identified (see Fig.3). This has been successfully interpreted as a new lattice airframe concept design (see Fig.4). To determine the geometric parameters of the lattice fuselage barrel for the best design configuration in terms of weight savings subject to stability, strength and strain requirements, parametric optimization of the lattice barrel is performed in this section.

The design for the detailed lattice structure is a multi-parameter optimisation problem, for which a metamodel-based optimization technique is used to obtain the optimal solution describing the lattice element geometry. The methodology for performing parametric optimization consists of five steps and they are:

i. Design of Experiments (DoE)

ii. Metamodel building by Genetic Programming (GP)

iii. Parameterized Finite Element Model

iv. Design variables and loads for parametric optimization

v. Parametric optimization of the fuselage barrel using Genetic Algorithm (GA)

These five steps are integrated to perform the multi-parameter optimisation of the lattice fuselage design and explained in the following sub-sections. In this paper, parametric optimization is applied for the detailed lattice design of a fuselage barrel by using Genetic Algorithms on global metamodels generated with Genetic Programming from a 101-point optimal Latin hypercube Design of Experiments.

3.1 Design of Experiments (DoE)

The quality of the metamodel strongly depends on an appropriate choice of the Design of Experiments (DoE) type and sampling size. To improve the quality of the conventional random Latin hypercube DoE, a uniform Latin hypercube DoE based on the use of the Audze-Eglais optimality criterion (Audze and Eglais 1977), is proposed. The main principles in this approach are as follows:

- The number of levels of factors (same for each factor) is equal to the number of experiments and for each level there is only one experiment;
- The points corresponding to the experiments are distributed as uniformly as possible in the domain of factors. There is a physical analogy of the Audze-Eglais optimality criterion with the minimum of potential energy of repulsive forces for a set of points of unit mass, if the magnitude of these repulsive forces is inversely proportional to the squared distance between the points:

$$U = \sum_{p=1}^{P} \sum_{q=p+1}^{P} \frac{1}{L_{pq}^{2}} \quad \rightarrow \min \tag{2}$$

where P is the number of points, L_{pq} is the distance between the points p and q ($p \neq q$) in the system. Minimizing U produces a system (DoE) where points are distributed as uniformly as possible. This approach has been further generalised to include a case of unequal number of levels in different design variables, referred to as an extended optimal Latin hypercube DoE (Toropov et al. 2007).

According this extended optimal Latin hypercube design of numerical experiments (DoE), a 101-point DoE has been developed for the FE simulation to be performed at each point. Since one of the design variables (the number of helical rib, in Section 3.4) was a discrete parameter within the bound between 50 and 150 (total 101 levels) and all the other design variables were continuous, that was why a 101-point Latin hypercube Design of Experiments was used in this paper.

The bar chart of the minimum distances between the sampling points is shown in Fig. 5 indicating a good uniformity of the DoE.

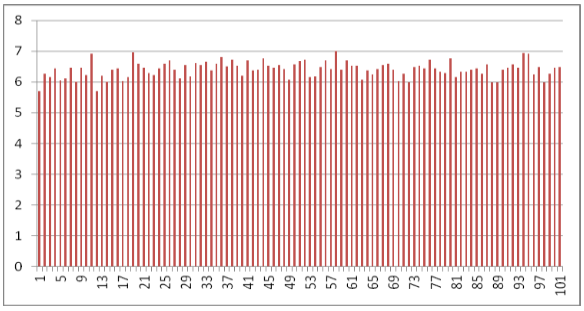


Fig. 5 Minimum distances between points in 101–point optimal Latin hypercube design of experiments

3.2 Metamodel Building by Genetic Programming (GP)

Genetic programming methodology (GP) (Armani et al. 2011, 2014; Koza 1992) is a systematic way of selecting a structure of high quality global approximations. Selection of the structure of an analytical approximation function is a problem of empirical model building. Selection of individual regression components in a model results in solving a combinatorial optimization problem. Even if the bank of all regressors is established (which is a difficult problem on its own), the search through all possible combinations would result in prohibitive computational effort. GP is based on the same basic methodology as genetic algorithms (GA). While a GA uses a string of numbers to represent the solution, the GP creates a population of computer programs of a certain structure. In our case of design optimization, the program represents an empirical model to be used for approximation of a response function.

These randomly generated programs are general and hierarchical, varying in size and shape. GP's main goal is to solve a problem by searching highly fit computer programs in the space of all possible programs that solve the problem. This aspect is the key to finding near global solutions by keeping many solutions that may potentially be close to minima (local or global). The creation of the initial population is a blind random search of the space defined by the problem. The evolution of the programs is performed through the action of the genetic operators and the evaluation of the fitness function. The common genetic operations used in genetic programming are reproduction, mutation and crossover, which are performed on mathematical expressions stripped of their corresponding numerical values. More details about genetic programming in this paper have been introduced by Armani (2014).

In order to encourage the evolution of smooth mathematical expressions and to avoid 'bloating' (Poli et al. 2008), the fitness values $F(i,t)$ of individual i at generation t has been defined as a weighted sum of different terms or objectives, following an approach used for multi-objective optimization in evolution-based algorithm:

$$F(i,t) = a_1 F_1(i,t) + a_2 F_2(i,t) + a_3 10^6 F_3(i,t) + a_4 F_4(i,t)\,, \tag{3}$$

$$a_1 + a_2 + a_3 + a_4 = 1 \tag{4}$$

where F_1 is the root mean square error (RMSE) of the ith individual in the tth generation evaluated on the given data set, divided by the average RMSE of the archive individuals at the previous generation; F_2 is the square of the number of numerical coefficients (parameters) present in the individual; F_3 is the number of operations not defined (i.e. division by zero) in the individual at any of the DoE sample point; F_4 is the number of nodes that the individual is made of and a_1, a_2, a_3 and a_4 are weighting factors (that add up to 1) determined by the exhaustive testing and tuning of the GP algorithm (Armani et al. 2011). Their values were: a_1=0.8989, a_2=0.001, a_3=0.1 and a_4=0.0001.

3.3 Parameterized Finite Element Model

101 finite-element analyses of composite lattice fuselage barrels corresponding to 101 DoE points are performed to generate the response sets for metamodel building. Detailed information about the FE modelling was given by Lohse-Busch (2013). The two FE models used in the analysis were based on a relatively coarse mesh and a much finer mesh that corresponds to a converged solution found from a mesh sensitivity study. The coarse mesh FE simulations, that are an order of magnitude faster, still reveal the most prominent features of the structural response. Then, the obtained optimal solution is validated by the analysis with the fine FE mesh.

3.3.1 Strength Normalization

In this paper, strain calculations by finite element analysis consisted of the tensile and compressive strains in the frames and helical ribs, and the tensile, compressive and shears strains in the fuselage skin. The strain criterion is the maximum allowable strain in the structure. Then, the margin of safety of the strains used as one of the constraints in the following optimization process is defined, which is a normalization with respect to the maximum allowable strain, see Eq. (5). It is a measure of whether the structures passes or fails due to the applied load. The strain margin of safety and hence strength normalised response is computed as

$$MS_{\varepsilon} = \frac{\varepsilon_{\max}}{\varepsilon} - 1 \geq 0 \tag{5}$$

where ε is the computed strain and $\varepsilon_{\max}$ the maximum allowable strain.

3.3.2 Stiffness Normalization

The margin of safety for stiffness (bending stiffness and torsional stiffness of the barrel structure) is computed as

$$MS_S = \frac{S}{S_{\min}} - 1 \geq 0 \tag{6}$$

where S is the computed stiffness and $S_{\min}$ the minimum allowable stiffness.

3.3.3 Stability Normalization

The margin of safety for buckling is computed as

$$MS_B = \lambda - 1 \geq 0 \tag{7}$$

where λ is the computed linear buckling eigenvalue for the applied loads.

3.3.4 Mass Normalization

The mass per unit meter was computed for each model and then normalized against the largest mass per unit length from the 101 DoE fuselage models.

Using the GP methodology and the 101 finite element generated response sets, metamodels for four different characteristics were built. These were the normalized responses of: 1) strength; 2) stiffness; 3) stability; and 4) the fuselage barrel mass. For the structural responses of strains, global stiffness and buckling, their margins of safety (MS) were calculated. These margins of safety can have either positive or negative values. A positive margin of safety shows that the computed value found in the structure does not violate the allowable value, and thus the structure is acceptable. A negative margin of safety, on the other hand, shows that the computed value violates the constraint defined by the allowable value. Hence the structure fails and should be redesigned. The normalization of the studied results allows for an easy comparison and a ready detection of failed fuselage geometries.

3.4 Design Variables and Loads for Parametric Optimization

In order to automate finite element models, the grid type fuselage section is a simple structure without windows or floors consisting only of the repeated structural triangular unit cell. Fig. 6 shows the finite element fuselage barrel model with the inner helical ribs in green, their counter parts on the outside of the skin in blue, the circumferential frames in yellow and the skin in red. The stiffening ribs are arranged at an angle so as to describe a helical path along the fuselage barrel skin. Hence, these ribs are called helical ribs. The helical ribs have a hat cross section, whereas the circumferential frames have a Z-shaped cross section. The upper barrel with the opaque skin illustrates the presence of only one set of parallel helical ribs on the outside of the skin. Below, the same barrel with transparent skin shows the presence of a second set of helical ribs winding around the barrel on the inside and in the opposite direction. These ribs in conjunction with the circumferential frames create uniform triangular skin bays. The helical ribs form an angle of 2φ between them as illustrated in Fig. 7. This angle remains constant throughout the barrel model.

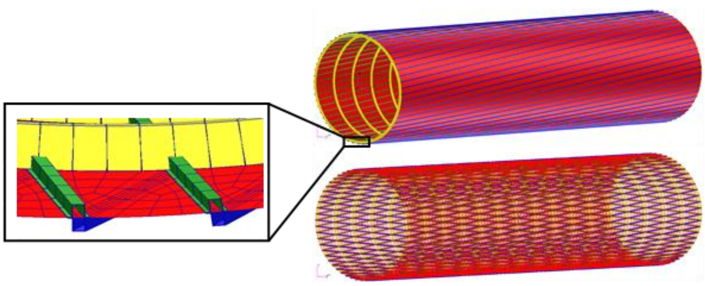


Fig. 6 Sample fuselage barrel finite element model

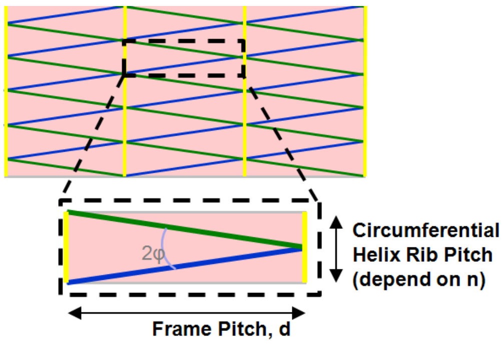


Fig. 7 Skin bay geometry showing the triangular skin bays.

The design variables are selected in order to vary the geometry of the helical stiffeners and frames, the skin thickness, and the frame pitch without altering the triangular shape of the skin bay geometry. The seven optimization variables (Fig. 7 and 8) are varied between the maximum and the minimum bounds listed in Table 1. By altering the frame pitch, the height of the triangular skin bay is affected. In Fig.7, we can conclude that the distance between two adjacent circumferential frames is equal to the length of the frame pitch and the longer the frame pitch, the smaller the number of circumferential frames. Similarly, the number of helical ribs changes the width of the base of these triangular bays. Consequently, these two variables change the area and the angle 2φ of these skin bays, and thus are, along with the skin thickness, instrumental in influencing the buckling behaviour of the structure. The rib and frame geometries are also affecting the buckling of the fuselage globally and locally. Stiffer reinforcements on the edges of the skin bays reduce global and skin bay buckling. Although stability is expected to be the critical failure mode for the fuselage structure, the variables are also affecting the strength of the structure. The composite material fails if it is strained beyond a maximum value. Finally, the fuselage has to have a certain stiffness in bending and in torsion to avoid excessive global deformations in flight.

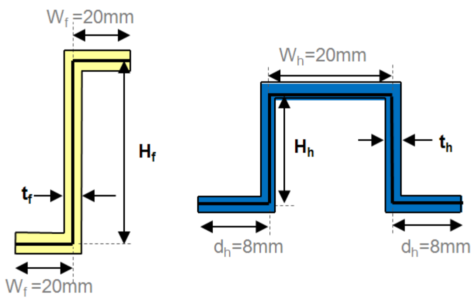


Fig.8. Geometry of the circumferential z-shaped rib and helical hat-shaped rib.

Table 1 Design variables

Design variables	**Lower bound**	**Upper Bound**
Skin thickness (h)	0.6 (mm)	4.0 (mm)
Number of helical rib pairs around the circumference, (n)	50	150
Helical rib thickness, (t_h)	0.6 (mm)	3.0 (mm)
Helical rib height, (H_h)	15 (mm)	30 (mm)
Frame pitch, (d)	500 (mm)	650 (mm)
Frame thickness, (t_f)	1.0 (mm)	4.0 (mm)
Frame height, (H_f)	50 (mm)	150 (mm)

An upward gust load case at low altitude and cruise speed is applied to the modelled fuselage barrel and depicted in Fig. 9. At one end of the barrel, bending, shear, and torsion loads are applied while the opposite end is fixed. These loads are applied via rigid multipoint constrains, which force a rigid barrel end. While floors are not modelled, the masses from the floors are applied at the floor insertion nodes. Finally, the structural masses are applied to the skin shell elements via mass densities.

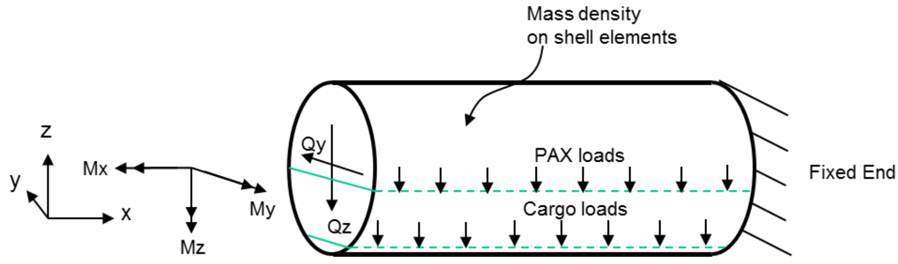


Fig. 9 Load application.

The optimization constraints are strain, global stiffness and stability. The corresponding optimization responses extracted from the FE models are the largest strains (tensile and compressive strains in the frames and in the helical ribs; tensile, compressive and shear strains in the skin), the critical buckling load, and the stiffness of the fuselage.

3.5 Parametric Optimization of the Fuselage Barrel Using Genetic Algorithm (GA)

The explicit expressions for the responses related to tensile strain, compressive strain, shear strain and weight of the fuselage barrel are built by GP. As an example, the expression for the shear strain is

$$\begin{aligned} f_{ss} &= 1.26902\, Z_1 - 1.76206\, Z_3 + 0.00132105\, Z_3\, Z_5 + 2.93847\, Z_3 / Z_2 + 603.316\, Z_3 / Z_5 + \\ &0.000000000604561\, Z_1\, Z_3\, Z_5^2\, Z_7 - 4143.98\, Z_1^2 /(Z_2^2\, Z_4\, Z_6) + 163.814\, Z_3\, Z_4 /(Z_2\, Z_5) + \\ &0.202164\, Z_2\, Z_4^2\, Z_6 /(Z_5\, Z_7^2) - 660.152\, Z_1^3 /(Z_2\, Z_3^2\, Z_4\, Z_7) - 15.5318\, Z_1^2\, Z_6\, Z_7^4 / \\ &(Z_2^4\, Z_3^2\, Z_4\, Z_5) - 0.975381 \end{aligned} \quad (8)$$

where Z_1 to Z_7 are the design variables detailed in Table 1. A graphical representation of the quality of the fit of the GP approximation for the shear strain response f_{ss} is shown in Fig. 10 where a point on the diagonal represents a perfect fit.

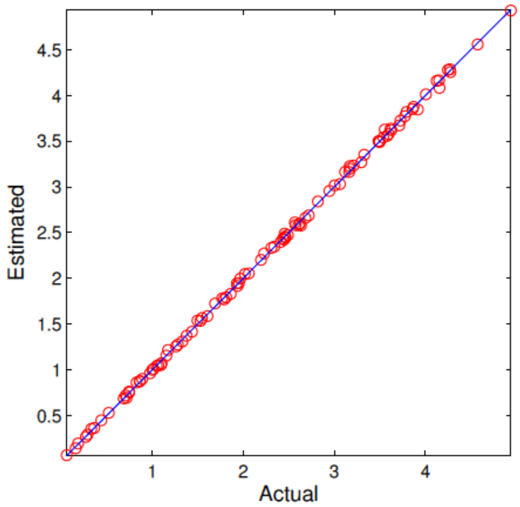


Fig. 10 Indications of the quality of fit of the obtained expression for the shear strain response into the data.

3.6 Results

The parametric optimization of GP-derived analytical metamodels for the fuselage barrel was performed by GA. Since GA has good non-local properties and is capable of solving problems with a mix of continuous and discrete design variables, it becomes a good choice for the fuselage barrel optimization where one of the design variables, the number of helical ribs, is integer. The results of the metamodel-based optimization and the fine mesh FE analysis are given in Table 2.

Table 2 Structural response values for the optimum design

Structural response type	Strain tension	Strain compression	Strain shear	Buckling	Torsional stiffness	Bending stiffness	Normalized mass
Prediction by the metamodel	0.20	0.23	1.27	0.00	1.21	0.89	0.29
Fine mesh FE analysis	0.62	0.08	1.09	-0.07	1.21	0.89	0.29

Results in Table 2 show that buckling is the driving criterion in obtaining the optimum. The metamodel-predicted optimum has a critical margin of buckling of 0.00 with a normalized weight of 0.29. However, when this was checked with a finite element analysis using a fine mesh, this value was found to be -0.07 that is unacceptable. This issue has to be addressed by interpreting the skin as a valid composite laminate, see Section 7.2. The predicted tensile strain margin of 0.20 is conservative when compared the 0.62 margin obtained by the FE analysis. The predicted compressive and shear strain of 0.23 and 1.27, respectively, are not conservative compared to the compressive strain margin of 0.08 and the shear margin of 1.09 obtained by the FE analysis. This is acceptable as these are not the critical margins. The predicted stiffness margins are the same as the margins obtained by the FE analysis but do not act as critical constraints in this design optimization problem. The design variable set for the final optimum geometry is listed in Table 3.

Table 3 Design variable values for optimum obtained with stability, global stiffness and strain constraints

Design variable	Skin thickness (h), mm	No. of helical rib pairs, (n)	Helical rib thickness, (t_h), mm	Helical rib height, (H_h), mm	Frame pitch, (d), mm	Frame thickness, (t_f), mm	Frame height, (H_f), mm
Optimum value	1.71	150.00	0.61	27.80	501.70	1.00	50.00

The length of the frame pitch is 501.7 mm which is close to the lower bound of 500. The resulting small triangular skin bays have a base width of 83.78 mm, a height of 501.7 mm and a small angle between the crossing helical ribs of $2\varphi=9.55°$. Such small and skinny-triangular skin bays are excellent against buckling. There is a good correspondence of the obtained results with the analytical estimates of DLR that produced the value of $2\varphi=12°$.

4. *Practical Design for a Laminated Skin*

The final phase of the proposed procedure is to generate a practical design of the skin component based on the optimum. In the above parametric optimization process, the smeared laminate properties have been assigned to the skin of the fuselage structure for the finite element analyses. These properties have the same effective laminate properties to represent the smeared layups. In this paper, the smeared laminate properties used in the FE simulations are: Young's modulus in the fiber direction E_1= 67 GPa; Young's modulus in the transverse direction E_2=55 GPa; Shear modulus G_{12}=25 GPa; Poisson's ratio γ_{12}=0.3; Material density ρ= 1600kg/m^3.

Since the optimal design only used smeared ply properties, the skin thicknesses had to be corrected to account for a standard CFRP ply thickness of 0.125 mm. This means that the skin thickness is increased from 1.71 mm to 1.75 mm (this means the normalized mass will be slightly larger than 0.29) and plies of 0°, 45°, -45° and 90° orientation arranged in a balanced and symmetric laminate have to be used to comply with the aircraft industry lay-up rules and manufacturing requirements (Kassapoglou 2013; Liu 2011; Niu 1992). The structural responses obtained by the FE analysis with the $(\pm45/90/45/0/-45/0)_s$ laminate skin are given in Table 4.

Table 4 Optimum response using 0.125 mm laminate plies

Structural response type	Strain tension	Strain compression	Strain shear	Buckling	Torsional stiffness	Bending stiffness	Normalized mass
Structural response margin of safety	1.15	0.19	1.31	0.13	1.25	0.81	Slightly larger than 0.29

Incorporating the ply thicknesses into the design has increased the buckling margin of safety making all margins positive. Therefore a light-weight design which fulfils the stability, global stiffness and strain requirements (Table 4) has been obtained. The ply angle percentages are as follows: 0 °plies: 28.6%, ±45 °plies: 57.1% and 90 °plies: 14.3%. A further study of a change of stacking sequence has been performed and reported in Table 5. It can be seen that moving the 0 °plies towards to the mid-plane of the laminate considerably improves the buckling performance of the skin.

Table 5 Effect of the skin stacking sequence on the structural responses

Stacking sequence	Buckling (margin of safety)
$(\pm45/0/45/0/-45/90)_s$	-0.04
$(\pm45/0/45/90/-45/0)_s$	0.04
$(\pm45/90/45/0/-45/0)_s$	0.13

5. Conclusions

A procedure for the implementation of composite lattice structures into the commercial aircraft was developed in this paper. The fuselage design was achieved by a three-stage process. First, topology optimization was performed to determine the conceptual design of a fuselage barrel section. The efficient arrangement of stiffeners was a concentration of stiffeners being included on the upper and lower extremities of the fuselage cross section and for these to form a criss-cross mesh pattern along the sides of the fuselage. Subsequently, parametric optimization was applied to the detailed design of a lattice fuselage barrel by using Genetic Algorithms on a metamodel generated with Genetic Programming from a 101-point optimal Latin hypercube Design of Experiments. The stability criterion was the driving factor for the skin bay size and the fuselage weight during the detailed lattice fuselage barrel design. The optimal solution and structural responses were verified with fine finite element simulations. In order to obtain a practical design of the laminated skin in the final stage, the skin modelled with smeared laminate properties has been successfully interpreted as a detailed composite laminate with a ply thickness of 0.125 mm to satisfy the aircraft industry lay-up rules and manufacturing requirements. It is concluded that the mixed optimization method, combining the use of topology optimization with the global metamodel-based approach, has allowed to solve the problem with sufficient accuracy as well as provided the designers with a wealth of information on the structural behaviour of the novel anisogrid composite fuselage design.

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