TOPOLOGY AND PARAMETRIC OPTIMIZATION OF A LATTICE COMPOSITE FUSELAGE STRUCTURE

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Abstract
Conventional commercial aircraft fuselages use all-aluminium semi-monocoque structures where the skin carries the external loads, the internal fuselage pressurization and is strengthened using frames and stringers. Environmental and economic issues force aircraft designers to minimize weight and costs to keep air transport competitive and safe. But as metal designs have reached a high degree of perfection, extraordinary weight and cost savings are unlikely in the future. Carbon composite materials combined with lattice structures and the use of topology optimization have the potential to offer such weight reductions. The EU FP7 project Advanced Lattice Structures for Composite Airframes (ALaSCA) was started to investigate this.

An anisogrid composite fuselage section was optimized using topology optimization with respect to weight and structural performance. The fuselage was parameterised and then a detail optimization was carried out using Genetic Algorithms on a metamodel generated with Genetic Programming from a 101 point Latin hypercube design of experiments. Two optimum lattice fuselage barrels were obtained and verified with finite element simulations. The first was optimized for strength requirement, producing a light weight fuselage with few thin helical ribs and circumferential frames, and large skin bays. The second was optimized for strength, stability and stiffness requirements, producing a heavier structure with smaller skin bays and more stiffeners, where stability became the driving
criterion. It is concluded that the use of the global metamodel-based approach combined with topology optimization has allowed to solve this optimization 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. This article presents this research which has now led to the development of a new airframe concept.

**Keywords** Metamodel; anisogrid structure; Latin hypercube; topology optimization; Genetic programming; aircraft fuselage design.

1. **Introduction**

Conventional commercial aircraft fuselages use all-aluminium semi-monocoque structures, which exploit the isotropic properties of the metal. The skin carries the external loads from the wings, empennage, engines, etc. as well as the internal pressurization of the fuselage. The skin is then strengthened using a series of frames and stringers which are either riveted or bonded [1]. These components are used to make large tubular sections, which are then joined with fasteners to form the complete fuselage.

Environmental and economic issues force future aircraft designs to maximize efficiencies in weight and cost to keep air transport competitive and safe. But as metal designs have reached a high degree of perfection, extraordinary weight and cost savings are unlikely to be achieved in the future [2].

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 % [3]. 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 [4]. The potential of composites has therefore not been completely realized using conventional aircraft airframe layouts [5].

In the 1980s, the composite lattice structure was developed by the Russian Central Research Institute for Special Machinery (CRISM) for rocket structures [6], Fig. 1 [7]. These structures consist of ribs either helically or ring shaped made of unidirectional composites fibres using automatic filament winding, also known as anisogrid structures. 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 high mechanical properties of the unidirectional composites of the lattice ribs are the main factor for their high weight efficiency. The automatic filament winding process ensures an integral structure with a low manufacturing cost. These structures are most efficient if they are undisturbed, (i.e. without cut-outs etc.) and at present are therefore predominantly applied in sections of space rockets [3].

![Fig. 1. A typical composite lattice structure without the external skin](image)

These experiences open up new opportunities for the optimization of composite aircraft fuselage barrels [2]. To determine this potential, a demonstrator was built as the DLR black fuselage concept [8, 9].

In order to address some of the issues associated with the implementation of composite lattice structure into commercial aircraft, the EU FP7 project titled Advanced Lattice Structures for Composite Airframes (ALaSCA) was started. This project consisted of carrying out a comprehensive investigation into the benefits of using the geodesic composite lattice structures used in the rocket industry and transferring the technology to the design of composite aircraft fuselages [10].

The aim of this article is to present the work carried out within the ALaSCA project which demonstrates that the combined use of topology and parametric optimization becomes an integral tool for the design of composite lattice aircraft fuselage barrel sections. The article presents: (1) the topology optimization problem, results and
their interpretation; (2) the different aspects of the primary composite lattice aircraft fuselage structure design, (3) parametric optimization of the lattice composite fuselage structure using a metamodel-based optimization technique, and (4) results and conclusions of this study.

2. Aircraft Configuration and Loads

This study used the aircraft configuration of the DLR funded project LamAiR, Fig. 2. The aim of that project was to increase the technology readiness level of the laminar airflow technology for short and medium range aircraft configurations [11].

The LamAiR aircraft configuration consists of a long fuselage barrel section mounted ahead of a forward swept wing and rear mounted turbine engines. Since the passenger and cargo doors (large cut-outs) are in the front cockpit-section or behind the wing, a long undisturbed barrel section can be designed as the main fuselage section. A composite lattice structure, together with the use of topology optimization are ideally suited for this type of configuration as it can lead to a light weight and cost efficient fuselage design. This configuration was therefore used in this study.

The loads for the fuselage barrel were analysed with the Preliminary Aircraft Design and Optimization (PrADO) program [12]. Four critical flight load cases were investigated and used to dimension the barrel. These load cases were: (1) two gust load cases (vertical and lateral) at low altitude; (2) one gust load case (vertical and lateral) at its maximum altitude and (3) a static internal pressure test load case whilst on the ground. Due to the rear position of the wing, this type of fuselage configuration experiences higher loads than an equivalent conventional swept back wing configuration

3. Methodology for the Topology and Parametric Optimization of Aircraft Structures

The process of combining topology optimization with parametric optimization to effectively design a composite lattice aircraft fuselage consists of six steps, these are:

1. Topology optimization
2. Topology extraction and simplification
3. Fuselage definition and design variable determination
4. Development of parameterized automatically generated finite element model
5. Generation of the metamodels
6. Optimization of the fuselage barrel using the metamodels

These eight stages in the new proposed process of aircraft fuselage design are explained in the next eight sections of this article.
4. Topology Optimization

The Solid Isotropic Material with Penalisation (SIMP) [13] topology optimization method was used to determine the fuselage stiffener optimal orientation. This was achieved using Altair’s OptiStruct program [14] which has its own finite element analysis solver.

The optimization problem consisted of minimizing the compliance of the fuselage section subject to the applied loads. This optimization problem is given by Equation (1). Note that the buckling requirements were not considered at this stage.

\[
\text{min } C_{\text{Tot}} \\
\text{subject to } \sum_{n=1}^{N} \rho_n V_n \leq V_0 \\
\rho_{\text{min}} \leq \rho \leq 1
\]

where: \( C_{\text{Tot}} \) is the total compliance for the structure, calculated using Equation (2) described in section 2.3; \( N \) is the total number of finite elements in the designable part of the structure; \( n \) is the number of the analysed finite element; \( \rho \) 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 \( \rho_{\text{min}} \) 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.

4.1. Fuselage section

The length of the fuselage section selected was 13,652 mm, it included a load and support introduction bays, both 399.8 mm in length. The rest of the fuselage had 23 bays with a 22” (558.8 mm) pitch, Fig. 3.

![Side view of the fuselage showing the different bays](image)

The cross section was made from three different radii and included a passenger and a cargo floor with struts connecting the two, Fig. 4.

Two different fuselage models were optimized, one with windows and one without. The model without windows used 25,584 shell elements, 5,842 beam elements and one RBE2 element [14], making it a total of 31,427 FE, Fig. 7. The model which included windows used 25,100 shell elements, 5,842 beam elements and one RBE2 element, making it a total of 30,943 FE, Fig. 8.

4.2. Applied loads

The loads applied were: a) distributed masses on the fuselage and its frames; b) point loads on the passenger and cargo floors; and c) point loads and moments at the free end of the fuselage barrel section.
There were five loads applied at the free end as three separate load cases, with the notation of Fig. 5. Load case 1 consisted of a vertical shear force $Q_z = 211,711$ N and moment $M_y = 446,965$ Nm. Load case 2 consisted of a torque about the $x$ axis applied in both directions, $T = \pm280$ kNm. Load case 3 consisted of a horizontal shear force $Q_y = \pm80$ kN and a bending moment $M_z = \mp249,614$ Nm applied in both directions. As part of Load case 1, a 3.47 load factor was applied acting in the negative $z$ direction on the mass of the fuselage and rings present at the bay locations. The fuselage mass of 2,461.9 kg was evenly distributed over the entire fuselage and the ring frames mass of 1322.8 kg was evenly distributed over the 23 rings. The passenger and cargo floor loads shown in Fig. 6 had the magnitudes: $F_{\text{Passenger}} = 3238.5$ N, $F_{\text{CC}} = 1510$ N and $F_{\text{CL}} = 755$ N.

4.3. Optimization parameters

The fuselage was optimized for each load case individually. The compliance values of the optimal topologies for the individual load cases were used to determine the multi-load weighting factors. These were used to calculate the multi-load case compliance value of Equation (2), which was used to optimize all load cases applied simultaneously. The logic behind this was that if the compliance value for the optimal topology of one of the load cases was higher than the others, that particular load case was critical. Hence it should have a higher
weighting in the total weighted compliance. The weighting factors for the fuselage without windows were: $\omega_1 = 0.4483$, $\omega_2 = 0.1604$, $\omega_3 = 0.3913$, and with windows were: $\omega_1 = 0.700$, $\omega_2 = 0.252$, $\omega_3 = 0.048$.

$$C_{Tot} = \sum_{Load=1}^{3} \omega_{Load}C_{Load} = \omega_1 C_1 + \omega_2 C_2 + \omega_3 C_3$$ (2)

4.4. **Optimal layout of stiffeners**

The fuselage section was optimised: 1) without windows, and 2) with windows. The results are given in Fig. 7 and Fig. 8 respectively.

5. **Topology Extraction and Simplification**

The results of Fig. 7 and 8 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 which criss-cross each other at different angles ranging from approximately 45° for fuselages with windows and approximately 60° for fuselages without windows. The ideal stiffener arrangement would therefore have continually varying angles, Fig. 9.

Such stiffeners are very complex and expensive to manufacture due to their continual varying sizes. As the fuselage would be manufactured using the pultrusion [15] process, stiffeners with constant cross-section and curvature are required. The design of Fig. 9 was simplified accordingly by using stiffeners of constant angle which were then aligned along geodesic lines on the fuselage skin, Fig. 10. This lattice structure generates non-rectangular skin bay segments in a grid-like arrangement which considerably improves the buckling resistance of the skin bays [17].

However, these grid stiffeners with very thin laminate thicknesses exhibit reduced axial stiffness when compared to conventional stiffened panels. This is primarily due to the aircraft industry requirements for repair and joining damage tolerance. To address this, a newly developed CFRP-metal-hybrid-material was used [18, 19]. The material has a high longitudinal stiffness because each CFRP unidirectional ply is bonded with a thin steel foil. The CFRP-metal hybrid is then stacked to produce the required laminate thickness. A penalty of using this material is that its density (1990 kg/m³) is 28% greater than conventional CFRP.
Fig. 7. Optimal stiffener layout for the fuselage without windows

Fig. 8. Optimal stiffener layout for the fuselage with windows
5.1. Selection of the grid stiffener configuration

Four grid stiffener concepts were analytically analysed using the barrel section dimensions of Fig. 3 with a 2.0 m radius circular fuselage cross-section using standard beam theory, composite laminate theory and linear buckling. The material used for the skin was Hexcel 8552/AS4 [16], it’s a CFRP with the following individual ply properties: Young’s modulus $E_{0°} = 141$ GPa, $E_{90°} = 10$ GPa, density $\rho = 1550$ kg/m$^3$.

The four concepts were: (1) A conventional semi-monocoque with CFRP stiffeners and 1.625 mm skin thickness; (2) A conventional semi-monocoque with CFRP-metal hybrid stiffeners and 1.625 mm skin thickness; (3) A grid arrangement with CFRP stiffeners and 1.625 mm skin thickness; and (4) A grid arrangement with CFRP-metal hybrid stiffeners and 2.75 mm skin thickness. The stringer and frame arrangement was the same for all four concepts, with only one design variable, the stringer cross-sectional area. The frame pitch used was 500 mm. The stringer pitch above the window line was 150 mm, giving a total of 24 stringers. The stringer pitch below the window line was 105 mm, giving a total of 81 stringers.

The only structural components used in the weight comparison were the fuselage skin, stiffeners and frames, Fig. 13. The results for each configuration were normalised with respect to the base concept 1. The combination of grid and CFRP-metal hybrid stiffeners gave the best weight reduction, totalling 20%.
Although the grid arrangements provide the best weight savings, they are also the most challenging to manufacture. If the stiffeners are placed on one side of the skin, at the location where the stiffeners cross, one of them needs to be cut to let the other one through. This reduces the structural integrity of the cut stiffeners and requires a significant weight penalty to remedy. For this reason, a new design was developed as part of this work. The stiffeners in one helical direction are placed on the inside of the skin, and those in the opposite direction are placed on the outside of the skin, Fig. 12. This has the consequence that the outside stiffeners must be covered by an aerodynamic skin and a filling material between the internal pressurized skin and the aerodynamic outside skin, Fig. 12. These two components were also added to the weight estimation calculation.
To determine the best concept provided, the additional weights of these new components (skin, foam, lightning strike material) were added to those of the primary structure and are given in Fig. 13. As for Fig. 11, the results were normalised with those of concept 1. The outer skin material for concepts 1 and 2 was a copper mesh with an area density of 370 g/m², and the thermal insulation was a mineral wool layer with an area density of 600 g/m². For concepts 3 and 4, outer skin was provided by a 0.2 mm aluminium layer with an area density of 540 g/m² and the thermal insulation was provided by the closed-cell foam Rohacell 31 IG with a Young’s modulus of 36 MPa, density of 31 kg/m³ and area density of 775 g/m² [20]. Concept 4 with the combination of grid and CFRP-metal hybrid stiffeners and aluminium light protection still gave the best overall weight reduction of 14.3%.

By implementing these findings, the design of Fig. 10 was re-evaluated and the new design concept for the ALaSCA fuselage barrel is given in Fig. 14.
6. Fuselage Definition and Design Variable Determination

The metamodel-based optimization technique (section 8) uses a response surface generated with results from full scale finite element analysis of the fuselage for a wide spectrum of values of the design variable. For each combination of design variable values, a separate finite element model is required. The process of generating these finite element models had to be automated; otherwise the cost of generating each model manually would make this an impossible process. Because of this, and since the purpose of this article is to demonstrate the automated optimization process, the fuselage barrel was further simplified from that given in Fig 14 by removing the windows, passenger and cargo floors and only concentrating on the repeated structural triangular cell units and the stiffeners.

Figure 15 shows a finite element model of the fuselage barrel 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 (called helical ribs) are arranged at an angle which describes a helical path along the fuselage barrel skin and have a hat-shaped cross section. The circumferential frames have a z-shaped cross section. The upper barrel with the opaque skin shows the presence of the outside parallel helical ribs. Below, the same barrel with transparent skin shows the presence of the second set of internal helix ribs wound in the opposite direction to the external ribs. These ribs in conjunction with the circumferential frames create uniform triangular skin
bays. The helical ribs form an angle of $2\phi$ between them, Fig. 16 below. This angle remains constant throughout the barrel model.

![Fig. 16. Skin bay geometry showing the triangular skin bays.](image)

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 design variables (Fig. 16 – 17) and their limits are:

1. $h$ is the skin thickness \(0.4 \leq h \leq 4.0\) mm
2. $n$ is the number of helical rib pairs around the circumference of the barrel \(50 \leq n \leq 150\)
3. $t_h$ is the helical rib thickness \(0.6 \leq t_h \leq 3.0\) mm
4. $H_h$ is the helical rib height \(15 \leq H_h \leq 30\) mm
5. $d$ is the circumferential frame pitch \(500 \leq d \leq 650\) mm
6. $t_f$ is the circumferential frame thickness \(1.0 \leq t_f \leq 4.0\) mm
7. $H_f$ is the circumferential frame height \(50 \leq H_f \leq 150\) mm

By altering the frame pitch, the height of the triangular skin bay is affected. Similarly, the number of helical ribs changes the width of the base of these triangular bays. Hence these two variables change the area and the angle $2\phi$ of these skin bays.

![Fig.17. Geometry of the circumferential z-shaped rib and helical hat-shaped rib.](image)

The fuselage structure is made from composite materials which are stacked laminated composite layers with different orientation angles. In order to considerably reduce the complexity of the optimization problem, the orthotropic properties of the composite plies were smeared into isotropic properties by calculating the average of $0^\circ/\pm45^\circ/90^\circ$ orientations. Once the optimum of the skin thickness was calculated, in order to account for the number of plies, the skin thickness was rounded up to the next integer number of plies.

### 7. Development of Parameterized Automatically Generated Finite Element Model

A tool was developed which allowed multi-parameter fuselage barrel finite element models to be generated automatically using the design variables specified in section 6. The tool simulates how a user generates a composite anisogrid fuselage barrel, its supports, and loadings, carry out the analysis and extract the relevant structural behaviour. The software used to carry out the finite element analysis was MSC Nastran and MSC
Patran [21] with the automatic model generation tool written using the Patran Control Language (PCL) [22]. The tool consisted of three separate parts: 1) a request file, 2) a pre-processing function and 3) a post processing function.

1. The request session file contains the list of parameters which are used as the input parameters for the model generation. This code contains the list of fuselage models to be simulated.
2. The pre-processing file generates the fuselage models, requests the analysis, and after the analysis is complete, calls the post processing function.
3. The post processing file extracts and formats the results.

A user defined number of models can be generated and analysed in batch mode by specifying the design variables from section 6 and the types of analysis to be carried out: linear elastic, buckling etc... The detailed flowchart for the automated multi-parameter global barrel finite element tool is shown in Fig. 18.

![Automated multi-parameter global barrel finite element tool flowchart](image)

The finite element model generated by this tool consists of the constant radius simplified fuselage barrel of Fig. 15 with triangular skin bays bounded by hat-shaped helical ribs and z-shaped circumferential frames. The skin is modelled with shell elements. The helical ribs, which are located on either side of the skin, are modelled with offset beam elements. The circumferential frames, located on the inner side of the skin, are also modelled with offset beam elements. A sample fuselage model is shown in Fig. 19 with a three dimensional visualization of the stiffeners modelled with beam elements.

### 7.1. Applied loads

The load applied consisted of an upward gust at low altitude and cruise speed. At the free end, bending moments, shear forces and torsion loads are applied as shown in Fig. 5. These loads were applied using rigid multipoint constrains, which force a rigid barrel end. In this simplified fuselage barrel model, the passenger and cargo floors were not directly modelled; but their masses were applied to the fuselage barrel where the floors contacted the barrel, Fig. 6.
8. Generation of the Metamodel

As seen from the definition of the design variables in Section 6, the number of helical rib pairs around the circumference of the barrel is an integer number that can vary between 50 and 150, and all other design variables are continuous. This makes a direct application of a fast gradient-based optimization technique linked to the finite element analysis impossible, and the use of one of the population-based techniques infeasible due to an excessive number of calls for the finite element analysis to perform. Therefore, a metamodel-assisted optimization approach has been chosen. The generation of the metamodel to represent the behaviour of the fuselage barrel consisted of two stages:

1. Design of Experiments (DOE) to generate a cloud of response points and

8.1. Design of experiments

A 101 point DOE using the uniform Latin hypercube sampling method was used in this work [23]. Each point of the DOE corresponded to a different set of design variable values and a separate finite element model of the barrel was generated and analysed to determine the performance of the fuselage barrel at each point.

8.2. Metamodel generation using GP

The Genetic Programming (GP) method [24] was used to generate the metamodels as it was found to be an efficient way of producing high quality explicit global approximations [25, 26]. In order to prevent discontinuities in the mathematical expressions and avoid creation of expressions of unnecessary complexity (bloating) [27], the fitness function values $F(i,t)$ given by Equation (3) for the $i^{th}$ individual in the $t^{th}$ generation was defined as a weighted sum of different objectives:

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

where $F_1$ is the root mean square error (RMSE) of the $i^{th}$ individual in the $t^{th}$ generation evaluated on the given data set, divided by the average RMSE of the population of individuals in 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 that are not defined (e.g. division by zero) in the individual evaluated at any of the DOE points; $F_4$ 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 [25]. Their values were: \( a_1=0.8989, a_2=0.001, a_3=0.1 \) and \( a_4=0.0001 \).

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 three structural responses of strains, 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 allowable value. Hence the structure fails and should be redesigned. The normalization of the studied results allows for an easy comparison and, in the case of the margins of safety, a ready detection of failed fuselage geometries.

8.2.1. Strength normalizing

The measure of strength used was the largest strains in the structure. This 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 margin of safety of the strains was then calculated. This is normalization with respect to the maximum allowable strain in the structure. 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 using Equation (4).

\[
MS_e = \frac{\varepsilon_{\text{max}}}{\varepsilon} - 1 \geq 0
\]  

where \( \varepsilon_{\text{max}} \) is the computed strain and \( \varepsilon \) the maximum allowable strain.

8.2.2. Stiffness normalizing

The margin of safety for stiffness is computed using Equation (5).

\[
MS_s = \frac{S}{S_{\text{min}}} - 1 \geq 0
\]  

where \( S \) is the computed stiffness and \( S_{\text{min}} \) the minimum allowable stiffness.

8.2.3. Stability normalizing

The margin of safety for buckling is computed using Equation (6).

\[
MS_b = \lambda - 1 \geq 0
\]  

where \( \lambda \) is the computed linear buckling eigenvalue for the applied loads.

8.2.4. Mass normalizing

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.

9. Optimization of the Fuselage Barrel Using the Metamodels

Two different optimizations were carried out of the fuselage barrel of Fig. 15. In case I, only the strength responses were used to generate the optimal fuselage geometry. This was done because the generation of the strength results could be generated a lot faster than the buckling results. In case II, all of the responses describes in section 8.2 were used to generate an optimal fuselage structure.

The optimum fuselage geometry predicted by the optimization of the metamodel was analysed by carrying out a detailed finite element analysis of it, and the two responses were compared. The optimization result was deemed acceptable if: 1) all margins of safety were positive; 2) the margins of safety predicted by the metamodel optimization were within 0.10 of the equivalent MS obtained from the detailed finite element model; and 3) the critical margin of safety from the predicted metamodel had a higher value than that of the detailed finite element.
analysis. When the result of the optimization on the metamodels obtained with the coarser FE mesh was checked against the FE analysis from the finer mesh, the difference in the critical response value was treated as a constant offset and added to the metamodel for correction, the optimization was repeated with the corrected metamodel until the requirements were met.

9.1. Optimization of Case I

For case I where only strength constraints were used, two optimization loops were required. Table I gives the responses predicted by the metamodel optimization (named Metamodel Prediction) together with the true responses from the finite element analysis of the structure generated with the optimized design variables. Two true responses were generated: 1) Using the parameterized automatically generated finite element tool from section 7, named (DOE mesh) and 2) Using a finite element mesh generated from a converged study based on the characteristics of design variables and the analysis results, named (Converged mesh).

<table>
<thead>
<tr>
<th>Model</th>
<th>Tensile strain (MS)</th>
<th>Compressive strain (MS)</th>
<th>Shear strain (MS)</th>
<th>Normalized mass</th>
</tr>
</thead>
<tbody>
<tr>
<td>Metamodel Prediction I</td>
<td>0.02</td>
<td>0.00</td>
<td>1.42</td>
<td>0.10</td>
</tr>
<tr>
<td>Optimum I (DOE mesh)</td>
<td>0.52</td>
<td>0.02</td>
<td>1.35</td>
<td>0.11</td>
</tr>
<tr>
<td>Optimum I (Converged mesh)</td>
<td>0.36</td>
<td>-0.09</td>
<td>1.21</td>
<td>0.11</td>
</tr>
<tr>
<td>Metamodel Prediction II</td>
<td>0.03</td>
<td>0.01</td>
<td>1.64</td>
<td>0.11</td>
</tr>
<tr>
<td>Optimum II (Converged Mesh)</td>
<td>0.54</td>
<td>0.04</td>
<td>1.54</td>
<td>0.12</td>
</tr>
</tbody>
</table>

The response predicted for the first optimum (Metamodel Prediction I) matches the DOE mesh response (Optimum I (DOE mesh)) better than the converged mesh response (Optimum I (Converged mesh)).

The first optimum has a normalized weight of 0.10 with a predicted critical margin of safety of 0.00 for the compressive strain, which for this problem is therefore the critical margin. In the finite element results for the DOE mesh, the critical compressive strain margin of safety is 0.02, with all other critical margin positive and hence giving conservative results. The optimization predicts accurately the critical structural response from a DOE generated with the automatically generated finite element tool. Although this is expected as the DOE was established with the same mesh. For the case of the converged mesh however, the critical margin has the negative value of -0.09, which indicates that the fuselage structure fails in compression. The strains obtained with the converged mesh have higher values than those obtained with the mesh generated with the parameterized automatically generated finite element tool. Due to this failure in the compressive strain, a second optimization loop was carried out. But due to the extra resolution of the converged mesh, instead of a requiring a zero margin of safety from the optimized metamodel, a positive margin was now allowed.

The second optimum (Metamodel Prediction II) had a slightly positive (0.01) and hence conservative critical margin of safety for the compressive strain. This time, the optimal design variables were only used to generate the model Optimum II (Converged mesh) using the converged mesh. The value obtained for the critical compressive strain was 0.04, which as it is less than 0.10 from the metamodel results it was deemed excellent. The tensile strain was the only criteria where the refined FEM result was greater than 0.10, however as the margin is positive and hence conservative, it was considered to not be important.

Table II shows the optimal values for the design variables for the two optimization runs. A weight efficient optimum was reached by minimising the number of frames and ribs and thus generating large skin bays. These are large triangular skin bays with a base width of 209.44 mm, a height of 627.70 mm and an angle between the crossing helical ribs of $2\varphi=18.94^\circ$. The helical ribs have a tall and slender hat-shaped cross section with a thickness of 0.66 mm, which is close to the minimal allowed valued of 0.60, and a height of 27.90 mm, also close to the maximum allowed value of 30 mm. This had led to a large moment of inertia and thus to a high bending stiffness. The circumferential frames, which are less instrumental in preventing fuselage bending have become thin and small, with both of their dimensions reaching the minimal bounds of 1.0 mm and 50.0 mm respectively. This case study has shown that when only strength responses are used in the optimization, the fuselage barrel is generated with large skin bays; few thin tall helical ribs; and few thin small circumferential frames.
9.2. Optimization of Case II

For case II, the three structural responses (strength, stability and stiffness) were used to generate the optimum. The optimum results from the metamodel optimization together with the true response from the converged finite element analysis are given in Table III.

Table III. Optimum obtained with strength, stiffness and stability constraints

<table>
<thead>
<tr>
<th>Model</th>
<th>Tensile strain (MS)</th>
<th>Compressive strain (MS)</th>
<th>Strain Shear (MS)</th>
<th>Buckling (MS)</th>
<th>Torsional Stiffness (MS)</th>
<th>Bending Stiffness (MS)</th>
<th>Normalized mass</th>
</tr>
</thead>
<tbody>
<tr>
<td>Metamodel Prediction III</td>
<td>0.20</td>
<td>0.23</td>
<td>1.27</td>
<td>0.00</td>
<td>1.21</td>
<td>0.89</td>
<td>0.29</td>
</tr>
<tr>
<td>Optimum III (Converged mesh)</td>
<td>0.62</td>
<td>0.08</td>
<td>1.09</td>
<td>-0.07</td>
<td>1.21</td>
<td>0.89</td>
<td>0.29</td>
</tr>
</tbody>
</table>

Unlike Case I, where the compressive strain was critical, when the full set of structural responses are incorporated, buckling becomes the critical driving criteria in obtaining the optimum. The metamodel predicted optimum has a critical margin in buckling of 0.00 with a normalized weight of 0.29. However, when this was checked with a finite element analysis using a converged mesh, this value was found to be -0.07. Although well within our 0.10 range of acceptable solution, it was nevertheless negative, therefore meaning that the fuselage would fail in buckling. Unfortunately due to time constraints, no additional optimization loops were able to be carried out.

Table IV shows the optimal values for the design variables for this optimization run. Stability requirements lead to smaller skin bays as larger panels buckle at a lower load than smaller panels. The skin bay area of the third optimum decreased by 68% compared to the skin bays of the second optimum. The number of helix ribs increase to 150, which is the upper bound for this variable. The frame pitch decreases to 501.50 mm, which is close to the lower bound of 500. The resulting skin bays are small triangular skin bays with a base width of 83.78 mm, a height of 501.70 mm and a shallow angle between the crossing helical ribs of $2\phi = 9.55^\circ$. These small and shallow skin bays are excellent against buckling. Also, the normalized weight increased considerably from 0.12 for the second optimum to 0.29 for the third optimum. The stability constraint was therefore found to be a weight driving factor.

Table IV. Design variable values for optimum obtained with strength, stiffness and stability constraints

<table>
<thead>
<tr>
<th>Design</th>
<th>$h$ (mm)</th>
<th>$n$</th>
<th>$t_i$ (mm)</th>
<th>$H_h$ (mm)</th>
<th>$d$ (mm)</th>
<th>$t_f$ (mm)</th>
<th>$H_f$ (mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Optimum III</td>
<td>1.71</td>
<td>150</td>
<td>0.61</td>
<td>27.80</td>
<td>501.70</td>
<td>1.00</td>
<td>50.00</td>
</tr>
</tbody>
</table>

The hat shaped helical ribs remain tall and thin with a thickness of 0.61 mm and a height of 27.80 mm. The z-shaped circumferential frames remain unchanged for the third optimum with a thickness of 1.0 mm and a height of 50.0 mm. These stiffeners are surprisingly thin and will suffer from local buckling, but as they were modelled using beam elements, then only global buckling could be considered in the optimization. This is something which was learned from this work, and in future all stiffeners must be modelled using shell elements in order to also capture local buckling effects. This of course would come at the expense of a much denser finite element mesh and hence computational time to obtain a result.

9.2.1. Inclusion of ply thicknesses into optimal design

Since the optimal design (Optimum III) 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 meant that the skin thickness increased from 1.71 mm to 1.75 mm. The new structural responses are given in Table V.

Incorporating the ply thicknesses has slightly increased the buckling margin of safety, although it is still negative with a value of -0.04. All other margins are positive. Considering the small negative buckling margin of
safety, it is expected that a small change such as an increase in skin thickness could be sufficient to obtain a zero or positive margin of safety. A preliminary light weight design, which fulfils the stability, strength and stiffness requirements, can be produced from this optimization result.

### Table V. Optimum III response using 0.125 mm laminate plies

<table>
<thead>
<tr>
<th>Model</th>
<th>Tensile strain (MS)</th>
<th>Compressive strain (MS)</th>
<th>Strain Shear (MS)</th>
<th>Buckling (MS)</th>
<th>Torsional Stiffness (MS)</th>
<th>Bending Stiffness (MS)</th>
<th>Normalized mass</th>
</tr>
</thead>
<tbody>
<tr>
<td>Optimum III (0.125mm ply)</td>
<td>1.15</td>
<td>0.19</td>
<td>1.31</td>
<td>-0.04</td>
<td>1.25</td>
<td>0.81</td>
<td>0.29</td>
</tr>
</tbody>
</table>

### 9.3. Comments on Results and Suggestions for Improvement

When only the strength constraints are used, the optimization results are valid and lead to a structurally sound preliminary fuselage design. By adding stability and stiffness constraints to the strength constraints, although the metamodel optimized result led to all MS being positive, the subsequent finite element analysis has led to a negative critical margin in the buckling constraint. Although in the current study this could not be improved to give all positive constraints, it is envisaged that the following two factors will address this issue: 1) an increase in the number of data points in the DOE; 2) the use of converged finite element mesh to determine the response of the structure at each of the DOE points. Because of time constraints this could not be implemented in this study.

### 10. Conclusions

Topology and parametric optimization was applied to the design of a fuselage barrel section. Topology optimization led to 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. This design was then interpreted, modified to incorporate manufacturing constraints and simplified to allow it to be parameterised in an automatic fashion.

The parameterised fuselage was then optimized further using Genetic Algorithms on a metamodel generated with Genetic Programming from a 101 point Latin hypercube design of experiments. The optimal solution and structural responses were verified with finite element simulations of the optimal lattice fuselage barrels. Two optimum structures were obtained. The first structure was optimized only for strength requirement, producing a light weigh fuselage with few thin helical ribs and circumferential frames, and large skin bays. The second structure was optimized for strength, stability and stiffness requirements. The fuselage generated was subsequently a heavier structure with smaller skin bays and more stiffeners. The stability criterion became the driving factor for the skin bay size and the fuselage weight. This optimum fuselage structure requires only small design change to produce an acceptable preliminary design. It is concluded that the use of the global metamodel-based approach combined with topology optimization has allowed to solve this optimization 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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### References


