ACCELERON AEROSPACE JOURNAL (AAJ)
E-ISSN: 2583-9942 (Online) | Accessible at www.acceleron.org.in
Volume 6, Issue 3, (pp.1765-1780)
Article ID: AAJ.11.2106-2613
Research Paper
https://doi.org/10.61359/11.2106-2613

High Fidelity Simulation Based Optimization of Aircraft Fuselage Structure

Mrugank Shukla[1*], Sagar Gupta[2†]

[1]Department of Aerospace and Autonomous Systems, Defence Institute of Advanced Technology, Pune, India.

[2]Department of Modelling and Simulation, Defence Institute of Advanced Technology, Pune, India.

Abstract: Aircraft Fuselage must be designed to achieve low structural weight while maintain adequate strength and stability under critical loading conditions. Conventional Global Finite Element Models (GFEM), typically composed of beam and shell elements, provide computational efficiency but may not capture detailed panel-level structural behavior. In this study, a high-fidelity finite element model of fuselage model is analyzed under Load Case to generate structural response data. A design of experiments (DOE) approach is used to explore the design space, and surrogate models are developed using polynomial regression for mass and box-cox regression for stress estimation. Structural Optimization is performed to minimize weight under strain constraints using Particle Swarm Optimization and brute-force search. The optimized configuration is further verified using a detailed finite element model (DFEM) of fuselage panel through eigenvalue buckling analysis, ensuring a buckling factor greater than 1.0.

Table of Contents

1. Introduction

The continuous demand for improved aircraft performance and reduced operating costs has driven the aerospace industry to develop lighter and more structurally efficient airframe designs. Among the primary load-carrying components of an aircraft, the fuselage plays a critical role in maintaining structural integrity while sustaining complex operational loads such as cabin pressurization, bending, shear forces, and local panel stresses. Consequently, structural design and optimization of fuselage structures have become important research topics aimed at improving structural efficiency while maintaining safety and reliability. In recent decades, advanced materials and improved structural design methodologies have significantly influenced aircraft structural development. Carbon fiber reinforced composite materials, for example, have been increasingly adopted in modern commercial aircraft and can account for more than half of the structural weight in aircraft such as the Boeing B787 and Airbus A350 (Kasapoglou, 2010). Compared with conventional metallic materials, composite structures provide advantages including higher stiffness-to-weight ratios, improved fatigue resistance, and reduced susceptibility to corrosion. These benefits enable the design of lighter and more integrated structural configurations (Kasapoglou, 2010).

Regardless of the material system, however, structural optimization of primary aircraft components generally focuses on the sizing of thin-walled structural elements such as fuselage skins, frames, and stiffeners. Key design parameters typically include variables such as skin thickness and stiffener dimensions, which strongly influence both structural performance and overall weight (Grihon et al., 2009). The design and optimization of large aerospace structures such as fuselages present considerable challenges due to the large number of design variables and the presence of multiple structural constraints. Typical failure mechanisms that govern fuselage structural design include stress limitations, stability requirements, and local buckling of thin-walled stiffened panels. Accurate evaluation of these structural behaviors often requires computationally intensive analysis methods. As a result, aircraft structural design commonly employs multi-level modelling approaches, where structural behavior is evaluated at different levels of fidelity to balance computational efficiency and modelling accuracy (Haftka & Gürdal,

*Department of Aerospace and Autonomous Systems, Defence Institute of Advanced Technology, Pune, India.
†Department of Modelling and Simulation, Defence Institute of Advanced Technology, Pune, India. **Corresponding Author:** sagarguptaoo63@gmail.com.
Article History: Received: 22-March-2026 || Revised: 28-March-2026 || Accepted: 28-March-2026 || Published Online: 31-March-2026.

1992). In such frameworks, global structural analyses are typically performed using relatively coarse finite element models that represent the entire fuselage structure using beam and shell elements. These models enable efficient evaluation of global load distributions and structural responses while allowing optimization algorithms to explore large design spaces (Grihon et al., 2009). However, simplified global models are not sufficient to capture detailed local phenomena such as panel buckling or local stress concentrations. For this reason, local analyses of fuselage structural components, such as stiffened skin panels or frame-bay structures, are often performed separately using more detailed structural models (Grihon et al., 2009).

Early efforts in fuselage structural optimization already recognized the importance of multi-level modelling strategies. One of the early optimization approaches proposed for fuselage structures combined simplified finite element models for global stress and deflection analysis with more detailed analytical formulations for evaluating structural constraints associated with frames, stiffeners, and skin panels (Sobieszczanski & Loendorf, 1972). Subsequent developments in aerospace structural optimization introduced more advanced multi-level and multidisciplinary optimization frameworks, in which complex design problems are decomposed into several interconnected optimization sub-problems (Sobieszczanski-Sobieski et al., 1985, 1987; Beers & Vanderplaats, 1987). In these approaches, information such as structural sensitivities and coupling parameters is exchanged between different levels of the optimization hierarchy in order to coordinate the overall design process (Barthelemy & Sobieszczanski-Sobieski, 1983). Although hierarchical decomposition techniques can effectively reduce computational complexity, some studies have shown that not all engineering systems can be represented strictly within hierarchical structures. Consequently, alternative optimization formulations have been proposed that rely on sensitivity-based coupling equations and non-hierarchical system representations (Sobieszczanski-Sobieski, 1988, 1990). In parallel with these developments, variable-complexity optimization methods have also been investigated, where models with different levels of fidelity are used within the same optimization process. Such approaches allow expensive high-fidelity simulations to be replaced or supplemented by lower-cost approximations during certain stages of the optimization procedure (Hutchison et al., 1994) (Burgee et al., 1996).

To further improve computational efficiency, researchers have increasingly employed surrogate modelling techniques to approximate the behaviour of complex structural models. Surrogate models, also referred to as meta-models or response surface models, are constructed using data obtained from a limited number of high-fidelity simulations and provide rapid predictions of structural responses for new design configurations. Various surrogate modelling techniques have been investigated in aerospace structural optimization, including polynomial response surfaces, kriging models, and artificial neural networks (Paiva et al., 2010) (Simpson et al., 2001). These models can significantly reduce computational cost while enabling efficient exploration of large design spaces.

In the context of fuselage structural optimization, surrogate modelling techniques have been used to represent structural responses such as panel buckling behaviour and stress distribution. For example, response surface approximations have been applied to represent the critical buckling load of composite stiffened panels, enabling efficient evaluation of stability constraints during the optimization process (Bettebghor, 2011) (Bettebghor & Bartoli, 2012). Although simplified analytical or semi-analytical tools are often used for such analyses, their applicability may be limited to specific structural configurations or loading conditions. As a result, more general finite element-based approaches are often required for accurate assessment of structural behaviour (Colson et al., 2007) (Fleury et al., 2010).

The increasing availability of computational resources and advanced finite element software has made it possible to incorporate high-fidelity structural analyses within the design process. Commercial analysis tools such as MSC NASTRAN and Abaqus enable detailed modelling of complex aerospace structures and are widely used for structural analysis and verification of aircraft components (MSC Software Corporation, n.d.). However, repeated use of high-fidelity simulations within optimization loops can still result in significant computational cost, which motivates the development of efficient surrogate-assisted optimization strategies.

In the present study, a simulation-driven framework is applied for the structural optimization of an aircraft fuselage structure. A high-fidelity finite element model of the fuselage is analysed under a critical loading condition corresponding to Load Case 9 (LC9). A Design of Experiments (DOE) approach is used to generate simulation data by varying key structural design parameters associated with the fuselage structure. Surrogate models are then developed to approximate the relationship between design variables and structural responses, where polynomial regression is employed to estimate structural mass and Box–Cox regression is used to predict stress behaviour. These surrogate models are integrated within an optimization framework aimed at minimizing structural weight while satisfying stress constraints using Particle Swarm Optimization (PSO) and brute-force search methods. Finally, the optimized configuration is validated through detailed panel-level finite element analysis, where eigenvalue buckling analysis is performed to ensure that the structural stability requirement is satisfied.

Acceleron Aerospace Journal || AAJ.11.2106-2613
Volume 6, Issue 3, pp (1765-1780)
E-ISSN- 2583-9942
https://dx.doi.org/10.61359/11.2106-2613

2. HIFI Structural Fuselage Model

A high-fidelity (HiFi) finite element model of a fuselage barrel was developed to evaluate the structural response of the fuselage under the specified loading conditions and to generate simulation data for the optimization study. The model represents a typical semi-monocoque aircraft fuselage configuration consisting of a thin cylindrical shell reinforced with longitudinal stringers and circumferential frames. Such stiffened shell configurations are widely used in aircraft fuselage structures because they provide high structural efficiency while maintaining low structural weight.

The fuselage geometry (Fig.1) considered in the present study has a radius of 2075 mm and consists of 20 circumferential frames along the longitudinal direction. The frame pitch is 584 mm, representing the distance between adjacent frames. This structural configuration represents a typical fuselage barrel section with sufficient structural length to capture load redistribution between neighbouring stiffened bays. The fuselage skin is reinforced by 70 longitudinal stringers uniformly distributed along the circumference of the fuselage. The stringers are assumed to have constant circumferential spacing corresponding to an angular spacing of approximately 4.706° between adjacent stringers. Uniform stringer spacing is commonly adopted in fuselage structural design because it ensures symmetric load transfer and consistent structural stiffness around the fuselage circumference (Grihon et al., 2009). The longitudinal stiffeners are modelled as hat-shaped stringers with parametric geometry. Hat stringers are widely used in fuselage structures because they provide favourable bending stiffness and efficient load-carrying capability while maintaining relatively low structural weight. The geometry of the hat stringer is defined using several parameters, including stringer height, stringer width, stringer top flange width, and stringer thickness. In addition to the stringer parameters, the skin thickness is also considered as a design variable in the structural model. These parameters directly influence both the stiffness of the fuselage shell and the local buckling behaviour of the stiffened panels.

Circumferential frames are modelled using a C-section cross-section with a frame width of 28 mm and frame height of 85 mm (Fig. 2). The frame thickness is assumed to remain constant at 1 mm. This assumption is introduced to reduce the dimensionality of the design space and to focus the optimization study primarily on the skin and stringer parameters, which have a stronger influence on fuselage structural weight. In preliminary fuselage structural design, frame dimensions are often fixed while skin and stiffener parameters are varied during sizing studies because they contribute more significantly to the structural mass and local stability characteristics of the fuselage shell.

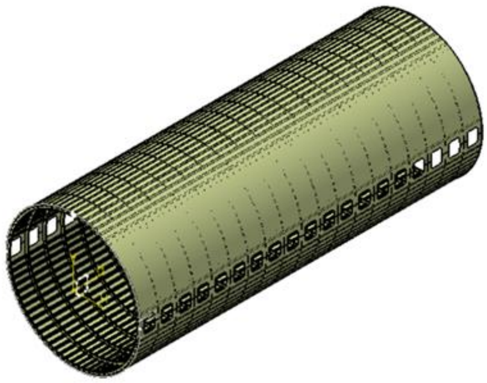


Figure.1

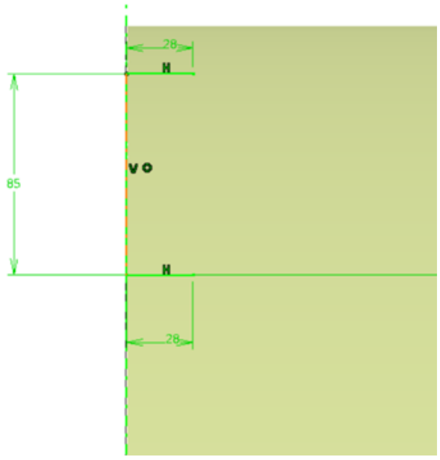


Figure.2

The fuselage structural components are assigned aluminium alloys commonly used in aircraft fuselage structures (Table 1). The fuselage skin is modelled using aluminium alloy Al2024-T3, which is widely used in aircraft skins due to its favourable fatigue resistance, good damage tolerance characteristics, and adequate strength under cyclic loading conditions (Bruhn, 1973). The stringers and frames are modelled using aluminium alloy Al7075-T6, which provides higher yield strength and stiffness compared to Al2024-T3. This higher strength makes Al7075-T6 particularly suitable for load-carrying structural members such as stiffeners and frames, which are responsible for resisting axial loads and maintaining the structural stability of the fuselage shell (Niu, 1988).

Table 1. Material Properties

Material Property	**AL 2024 T3**	**AL 7075 T6**
Density	2700 kg/m^3	2810 kg/m^3
Young's Modulus	73100 MPa	71700 MPa
Poisson's ratio	0.33	0.33
Yield Strength	385 MPa	450 MPa

The geometric design variables considered in this study are defined within the following ranges (Fig. 3):

- Stringer height: 15 – 30 mm
- Stringer width: 15 – 30 mm
- Stringer top flange width: 15 – 30 mm
- Skin thickness: 1 – 4 mm
- Stringer thickness: 1 – 3 mm

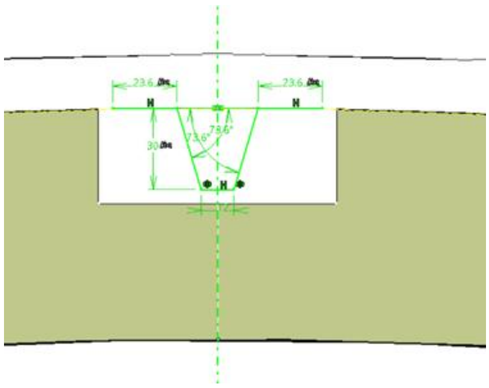
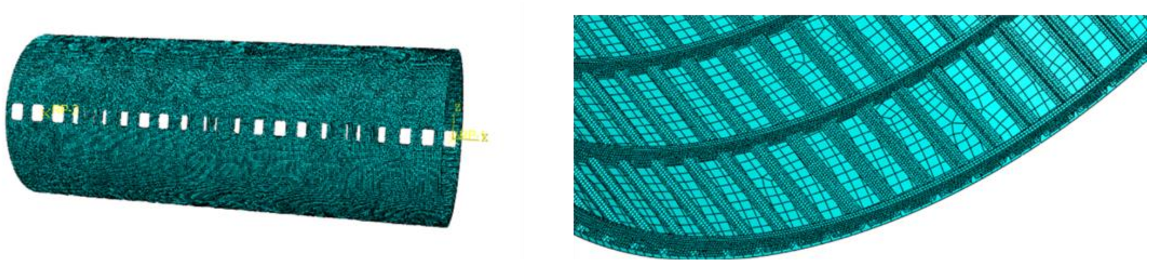


Figure. 3

These parameter ranges were selected based on typical dimensional limits used in aircraft fuselage stiffened panel design and to ensure that the resulting structural configurations remain within realistic manufacturing and structural design limits. Openings representing fuselage windows are also included in the structural model to capture their influence on the stress distribution within the fuselage shell. Window cut-outs introduce local stress concentrations and may significantly influence the load redistribution in the fuselage skin, making them an important feature to consider when evaluating structural behaviour.

All structural components in the fuselage model are represented using four-node reduced integration shell elements (S4R) available in the Abaqus finite element software package (Dassault Systèmes, n.d.). Shell elements are well suited for modelling thin-walled aerospace structures because they efficiently capture both membrane and bending behaviour while maintaining relatively low computational cost. Different mesh densities are adopted for different structural components to ensure adequate numerical accuracy. A mesh size of 50 mm is used for the fuselage skin, while a finer mesh size of 15 mm is applied to the stringers and frames (Fig. 4) in order to accurately capture stress gradients and local structural behaviour in the stiffening members.



Acceleron Aerospace Journal || AAJ.11.2106-2613
Volume 6, Issue 3, pp (1765-1780)
E-ISSN- 2583-9942
https://dx.doi.org/10.61359/11.2106-2613

Figure. 4

The objective of the structural optimization study is to minimize the total structural mass of the fuselage barrel while satisfying structural performance constraints. The structural constraints in the optimization problem are defined in terms of allowable Von Mises stress limits within the fuselage structure. In the present study, the maximum allowable Von Mises stress is constrained to lie within the range of 40–50 MPa. This range was selected to ensure that the stresses in the fuselage skin and stiffening members remain significantly below the material yield strength while maintaining structural efficiency.

For the aluminium alloys used in this study, namely Al2024-T3 for the skin and Al7075-T6 for the stringers and frames, the yield strengths are substantially higher than the imposed stress limits. By restricting the operating stress to the range of 40–50 MPa, sufficient safety margins are maintained against yielding, fatigue damage, and local structural instability in the thin-walled fuselage components. Such conservative stress limits are commonly adopted during preliminary aircraft structural sizing to ensure reliable long-term structural performance while enabling effective weight optimization (Bruhn, 1973, Niu, 1988). The resulting high-fidelity fuselage model provides a detailed representation of the structural configuration and forms the basis for generating simulation data used in the surrogate modelling and optimization framework developed in the present study.

3. HIFI Barrel Model Load Case and Analysis

Aircraft fuselage structures are subjected to multiple operational load conditions that arise from aerodynamic forces, cabin pressurization, manoeuvre loads, and structural interactions with other aircraft components. During the aircraft structural design process, a large number of load cases are evaluated to ensure that the fuselage structure satisfies strength and stability requirements under all critical operating conditions. In the present study, Load Case 9 (LC9) (Fig. 5) was selected from a total set of 15 predefined load cases because it produced the most significant stress levels within the fuselage barrel region. Selecting the most critical load case is a common approach in structural optimization studies since it ensures that the optimized configuration satisfies structural constraints under the governing loading condition.

Table 2

Fx	**2560 N**
Fy	1.4e5 N
Fz	1.54e-3 N
Mx	-22 Nmm
My	-13 Nmm
Mz	-1.26e8 Nmm

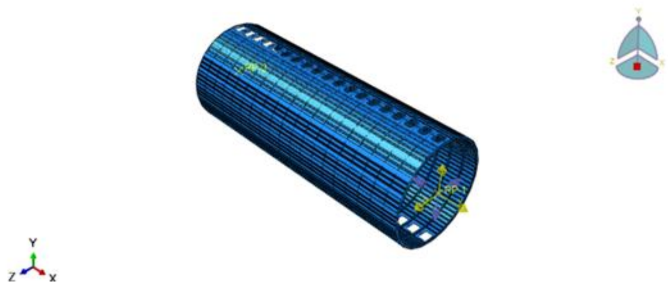


Figure .5

These loads represent the combined force and moment resultants acting on the fuselage barrel section during the selected operating condition. The structural analysis of the HiFi fuselage barrel model was performed using the Abaqus finite element solver (Dassault Systèmes, n.d.). A linear static analysis was conducted in order to evaluate the stress distribution and structural response of the fuselage under LC9 loading conditions. The loads were applied using a kinematic coupling constraint, where the resultant forces and moments were distributed across the nodes of the fuselage end section through a reference point. This approach ensures that the applied loads are transferred uniformly across the cross-section while preventing unrealistic local stress concentrations that may arise from direct nodal loading.

One end of the fuselage barrel was constrained using clamped boundary conditions, preventing all translational and rotational degrees of freedom. The loads were applied at the opposite end of the fuselage through the kinematic coupling reference point. This boundary condition configuration allows the fuselage section to develop realistic stress and deformation patterns under the applied loads. The structural connections between the fuselage components were modelled using tie constraints between the skin, stringers, and frames. Tie constraints enforce displacement compatibility between connected surfaces and ensure that the structural components behave as a single integrated structure during the analysis.

The linear static simulation was performed using four processor cores, resulting in a computational time of approximately 250 seconds for a single high-fidelity analysis. Before performing the static analysis, a modal analysis was conducted to verify that all structural components in the model were properly connected through the tie constraints. Modal analysis is commonly used as a verification step in finite element modelling because it helps detect unintended structural discontinuities or disconnected components within the model.

In a correctly connected structural system, the first six eigenmodes correspond to rigid body modes, which represent the unconstrained translational and rotational motion of the entire structure without any internal deformation. The presence of six rigid body modes (Fig. 6) therefore confirms that the structural components are properly connected and that no unintended degrees of freedom exist between the skin, stringers, and frames. The modal analysis results for the HiFi fuselage model show that the first six modes correspond to rigid body motions, indicating that the structural connectivity between the fuselage skin, stringers, and frames is correctly established through the tie constraints. This verification step ensures that the subsequent structural analysis results accurately represent the behaviour of the integrated fuselage structure.

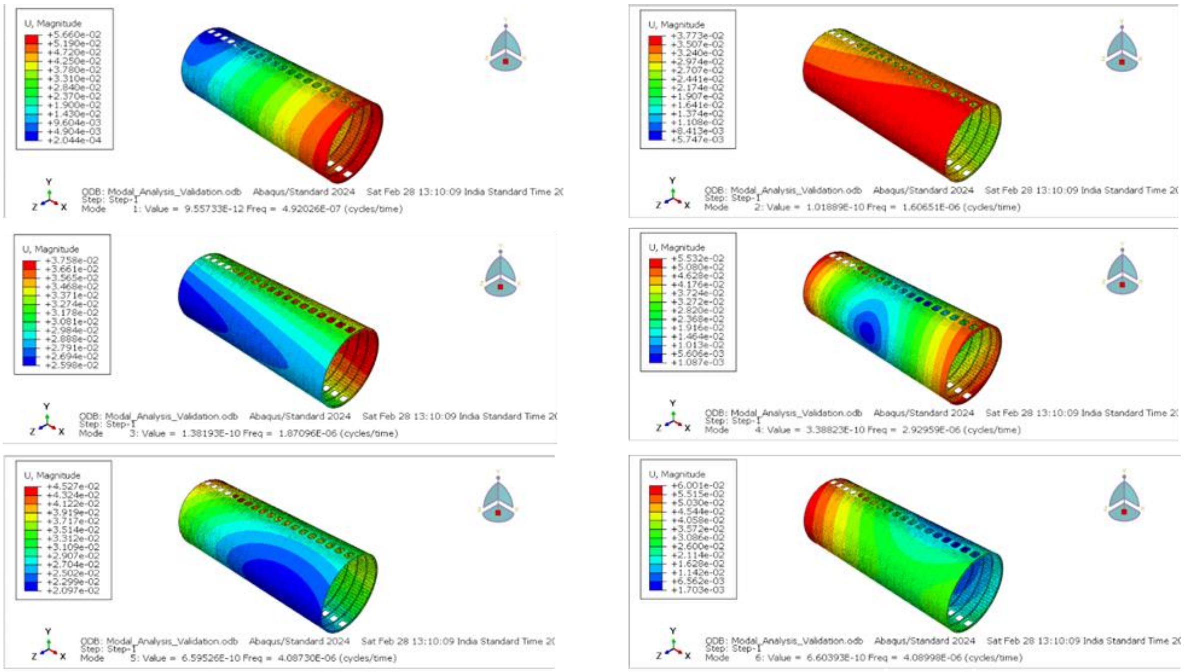


Figure 6. Six-Body Rigid Modes

To illustrate the structural behaviour of the HiFi fuselage model under LC9 loading, a reference structural configuration was analysed with the following geometric parameters:

- Stringer height: 30 mm
- Stringer width: 23.5 mm
- Stringer top flange width: 12 mm
- Stringer thickness: 1 mm

- Skin thickness: 1.6 mm

For this configuration, the linear static analysis resulted in a maximum Von Mises stress (Fig. 7) of approximately 61.83 MPa within the fuselage structure. This stress level remains significantly below the yield strength of the aluminium alloys used in the structural model, indicating that the selected configuration operates within the elastic structural regime under the considered load case.

The obtained stress distribution also highlights the interaction between the fuselage skin and stiffening members, demonstrating the effectiveness of the stiffened shell configuration in distributing loads throughout the fuselage structure. The results from this HiFi structural analysis serve as the baseline data for the subsequent design space exploration, surrogate modelling, and structural optimization procedures presented in the following sections.

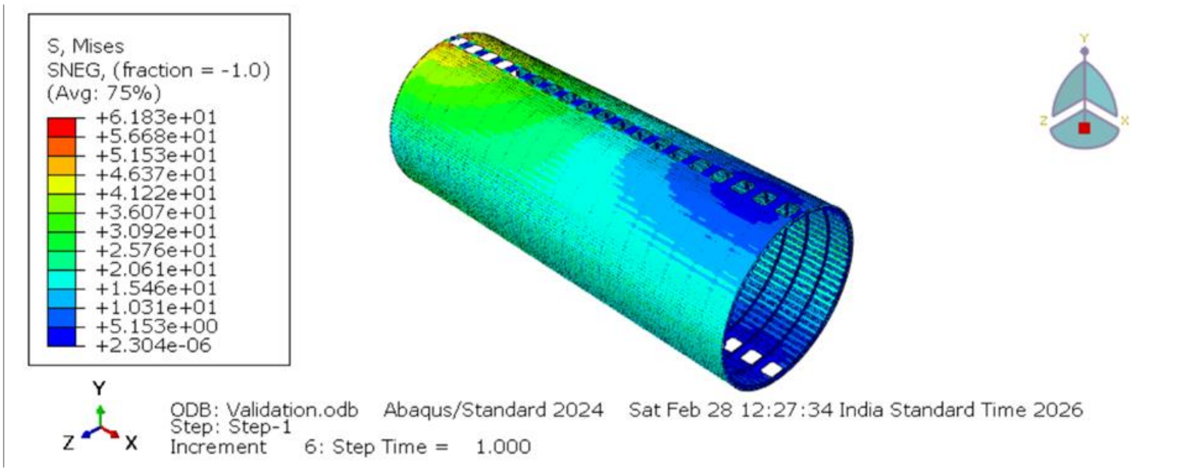


Figure 7. Von Mises stress

4. Design of Experiment (DOE) and Data Generation

In order to explore the influence of the geometric design variables on the structural performance of the fuselage, a Design of Experiments (DOE) approach was adopted. DOE techniques are widely used in engineering optimization problems to systematically sample the design space and generate representative datasets for surrogate modelling and optimization studies (Simpson et al., 2001).

In the present study, five geometric design variables associated with the fuselage stiffened panel configuration were considered:

- Stringer height
- Stringer width
- Stringer top flange width
- Stringer thickness
- Skin thickness

The ranges of these parameters were defined based on practical fuselage structural design limits and are summarized in Table 3.

Table 3 Structural Design Parameters

Design Variable	Range
Stringer height	15 – 30 mm
Stringer width	15 – 30 mm

Stringer top flange width	15 – 30 mm
Skin thickness	1 – 4 mm
Stringer thickness	1 – 3 mm

To efficiently sample this five-dimensional design space, a Sobol sequence sampling method was used to generate the DOE points (Fig. 8). Sobol sequences belong to the class of low-discrepancy quasi-random sequences, which are commonly used in design exploration because they provide uniform coverage of the design space while avoiding clustering of sample points (Simpson et al., 2001). Compared with purely random sampling methods, Sobol sequences provide better distribution of points across high-dimensional design spaces, which improves the quality of surrogate models trained on the resulting dataset. Using this approach, 100 design points were generated within the specified parameter ranges. Each DOE point represents a unique combination of the five geometric parameters, which was subsequently used to generate a corresponding high-fidelity fuselage model for structural analysis.

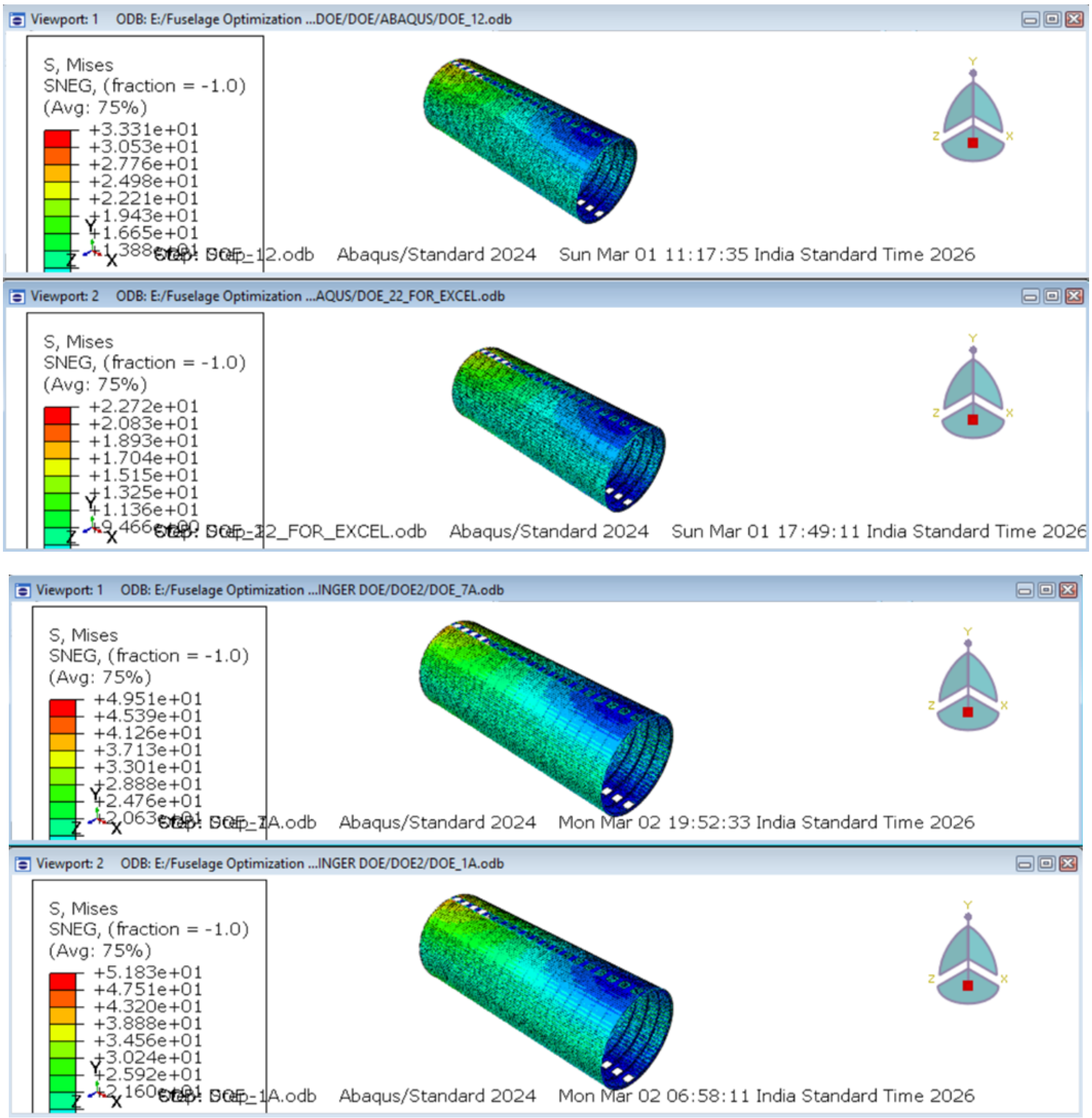


Figure 8. Von Mises Stress Analysis

5. Surrogate Modelling

The predictive performance of these models was assessed using the coefficient of determination (R^2) and the Root Mean Square Error (RMSE). The R^2 metric indicates how well the regression model explains the variance in the data, while RMSE measures the average magnitude of prediction errors.

Acceleron Aerospace Journal || AAJ.11.2106-2613
Volume 6, Issue 3, pp (1765-1780)
E-ISSN- 2583-9942
https://dx.doi.org/10.61359/11.2106-2613

The performance comparison of the three models is summarized in Table 4.

Table 4 Performance Comparison

Model	R^2	RMSE
Linear Regression	0.9898	48.506
Polynomial Regression (Degree 2)	0.9999	0.08285
Box–Cox Transformation + Regression	0.999	2.29324

From the results, it can be observed that the second-order polynomial regression model provides the highest prediction accuracy, achieving an R^2 value of 0.9999 and the lowest RMSE of 0.08285. This indicates that the polynomial model is capable of accurately capturing the relationship between the geometric design variables and the resulting fuselage mass. The linear regression model, although showing a relatively high R^2, produced a significantly larger RMSE, indicating reduced prediction accuracy. The Box–Cox regression approach improved the model performance compared to linear regression but still resulted in higher prediction errors than the polynomial model.

Based on these results, the second-order polynomial regression model was selected as the surrogate model for mass prediction in the subsequent optimization process. In addition to the mass model, surrogate models were also developed to predict the maximum Von Mises stress obtained from the HiFi finite element simulations. Predicting stress behaviour is generally more challenging than predicting mass because the stress response depends on complex structural interactions between the fuselage skin, stringers, and frames, leading to stronger nonlinear relationships between the design variables and the resulting structural response.

Similar to the mass modelling study, three regression approaches were evaluated:

- Linear Regression
- Polynomial Regression (Degree 2)
- Box–Cox Transformation followed by Regression

The performance of the models was evaluated using the coefficient of determination (R^2) and the Root Mean Square Error (RMSE). The results are summarized in Table 5.

Table 5 Results of Performance Analysis

Model	R^2	RMSE
Linear Regression	0.6929	11.2273
Polynomial Regression (Degree 2)	0.9856	2.42655
Box–Cox Transformation	0.999	0.0003912

The results indicate that the linear regression model performs poorly for predicting stress, with an R^2 value of 0.6929, which suggests that a simple linear relationship cannot adequately represent the structural response of the fuselage model. The second-order polynomial regression model significantly improves prediction accuracy, achieving an R^2 value of 0.9856 and reducing the RMSE. This improvement reflects the nonlinear nature of the stress response with respect to the geometric design variables. The Box–Cox transformation combined with regression provides the highest prediction accuracy, achieving an R^2 value of 0.999 and the lowest RMSE of 0.0003912. The Box–Cox transformation helps stabilize the variance and improves the linearity of the relationship between the design variables and the transformed stress response, allowing the regression model to capture the underlying structural behaviour more effectively.

Based on these results, the Box–Cox regression model was selected as the surrogate model for predicting the maximum Von Mises stress during the optimization stage. Table 6 presents the comparison between the actual fuselage mass (kg) obtained from HiFi simulations and the values predicted using the three regression models.

Table 6. Comparison of Fuselage Mass

Sample Case	Actual Mass	Linear	Polynomial	Box–Cox
1	1404.17	1405.66	1404.18	1394.08
2	969.82	977.40	969.74	992.14
3	1919.24	1916.28	1919.28	1915.14
4	1065.00	1069.79	1065.04	1073.29

A similar comparison was carried out for the maximum Von Mises stress (MPa) predictions, as summarized in Table 7.

Table 7. Comparison of Von Mises Stress Analysis

Sample Case	Actual Stress	Linear	Polynomial	Box–Cox
1	40.18	44.06	39.19	39.77
2	61.83	57.91	61.51	59.94
3	28.29	26.96	16.01	28.55
4	57.66	58.03	62.16	57.91

6. Structural Optimization

Particle Swarm Optimization (PSO) was employed to search for the optimal design configuration using the surrogate models. PSO is a population-based stochastic optimization algorithm inspired by swarm intelligence, where candidate solutions move through the design space based on their individual experience and the global best solution found by the swarm. The advantage of PSO in this study is that it allows efficient exploration of the design space while requiring significantly fewer function evaluations compared with exhaustive search methods. Since the structural responses are evaluated using surrogate models rather than expensive finite element simulations, the PSO algorithm can rapidly identify near-optimal design solutions.

The optimal design obtained using PSO is presented in Table 8 and Table 9.

Table 8 – Optimal Design Obtained Using PSO

Parameter	Value
Stringer Height (mm)	26.192
Stringer Width (mm)	20.149
Bottom Flange Width (mm)	15.000
Stringer Thickness (mm)	2.119
Skin Thickness (mm)	1.649

Table 9 - Predicted Structural Response

Response	Value
Minimum Mass	1222.822
Von Mises Stress	50.016 MPa

To validate the optimal solution obtained from PSO, an additional optimization was performed using a brute force search approach. In this method, the design space is systematically explored by evaluating all possible combinations of design variables within the specified bounds using a predefined step size of 0.1.

Although brute force search provides a reliable way to identify the global optimum, it is computationally expensive because the number of evaluations increases exponentially with the number of design variables. Therefore, such methods are typically used only for verification purposes rather than for practical large-scale optimization problems.

Acceleron Aerospace Journal || AAJ.11.2106-2613
Volume 6, Issue 3, pp (1765-1780)
E-ISSN- 2583-9942
https://dx.doi.org/10.61359/11.2106-2613

The optimal design obtained using brute force search is shown in Table 10 and Table 11.

Table 10 – Optimal Design Obtained Using Brute Force Search

Parameter	Value
Stringer Height (mm)	29.900
Stringer Width (mm)	20.200
Bottom Flange Width (mm)	18.700
Stringer Thickness (mm)	2.100
Skin Thickness (mm)	1.600

Table 11 - Predicted Structural Response

Response	Value
Minimum Mass	1250.008
Von Mises Stress	49.993 MPa

The results indicate that both optimization methods converge toward similar regions of the design space, confirming the validity of the surrogate-assisted optimization framework. The PSO algorithm achieved a slightly lower mass while maintaining similar stress levels, demonstrating the effectiveness of the algorithm in efficiently identifying optimal design solutions.

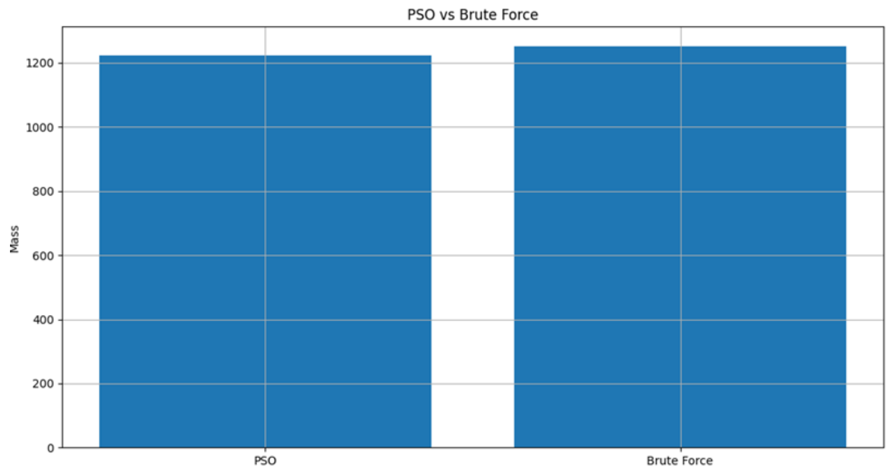


Figure 10. PSO Vs Brutal Force

Once the optimal design parameters were obtained from the surrogate-based optimization procedure, a high-fidelity finite element analysis (FEA) was performed to verify the predicted structural response. The optimized geometric parameters were applied to the HiFi fuselage finite element model, and the structure was analysed again under Load Case 9 (LC9) using Abaqus (Fig. 11). The objective of this step was to confirm that the surrogate model predictions were consistent with the results obtained from the high-fidelity structural simulation.

The comparison between the surrogate predictions and the FEA results provides an important validation step, ensuring that the optimization results remain accurate when evaluated using the detailed finite element model.

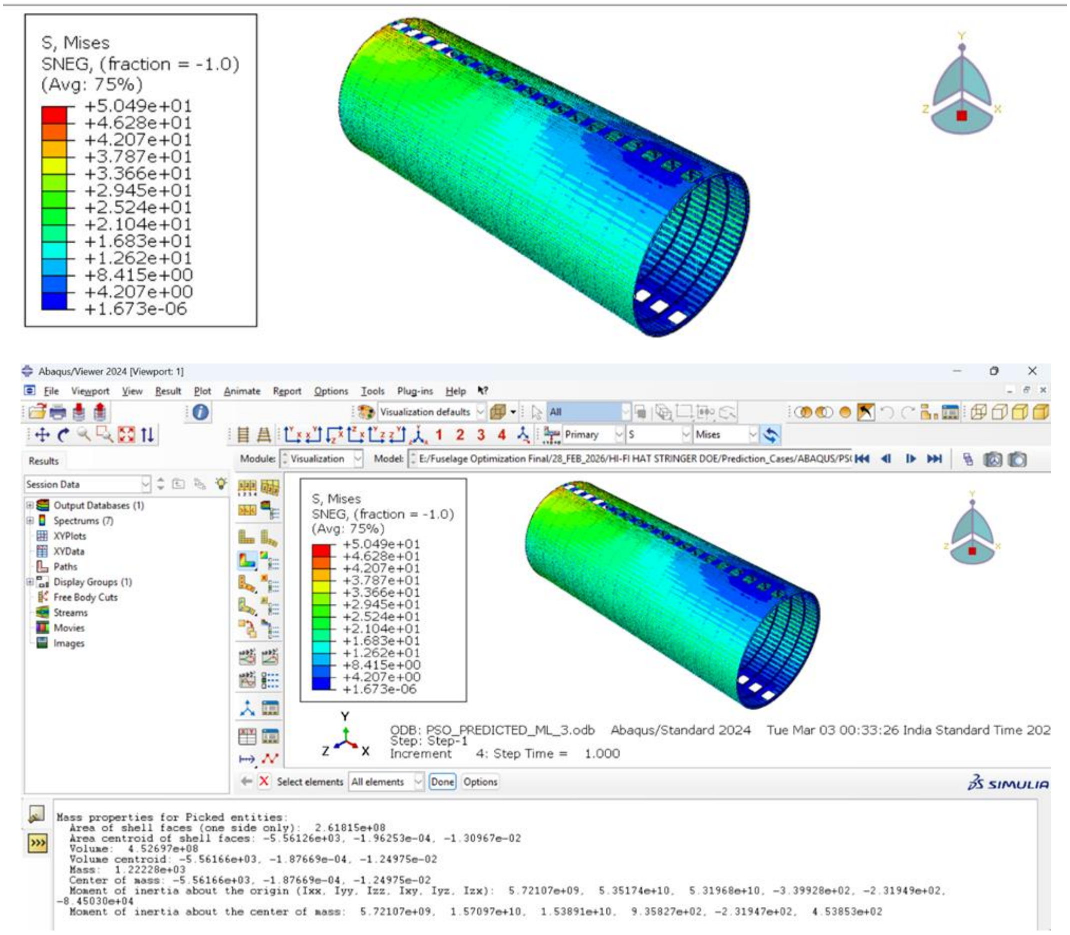


Figure 11. Analysed Under Load Case 9 (LC9) using Abaqus

Following the verification of the optimized fuselage configuration using the HiFi finite element model, section forces were extracted from the fuselage barrel model in order to perform a detailed structural assessment at the panel level.

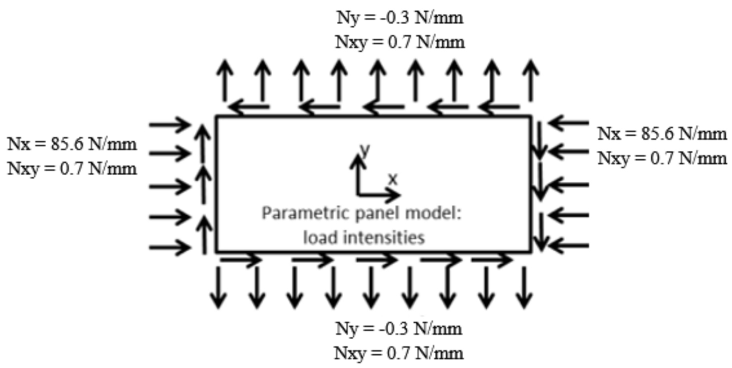


Figure 12. Parametric Panel Model

The extracted section forces represent the internal loads acting on a representative stiffened fuselage panel and are subsequently used as input boundary conditions for the Detailed Finite Element Model (DFEM) of the fuselage panel. This DFEM model allows accurate evaluation of local structural behaviour such as panel buckling and stability characteristics, which cannot be fully captured at the global fuselage modelling level. These quantities represent the in-plane membrane forces per unit length acting on the fuselage panel. The load distribution applied to the panel model is illustrated in Figure 12, where the extracted load intensities are applied along the panel edges.

Acceleron Aerospace Journal || AAJ.11.2106-2613
Volume 6, Issue 3, pp (1765-1780)
E-ISSN- 2583-9942
https://dx.doi.org/10.61359/11.2106-2613

7. Detail Panel Model

The detailed finite element model (DFEM) represents a local stiffened fuselage panel extracted from the global fuselage structure. In order to accurately capture the interaction between the fuselage skin, longitudinal stiffeners, and circumferential frames, the panel model includes five longitudinal stringers and four circumferential frames (Fig. 13).

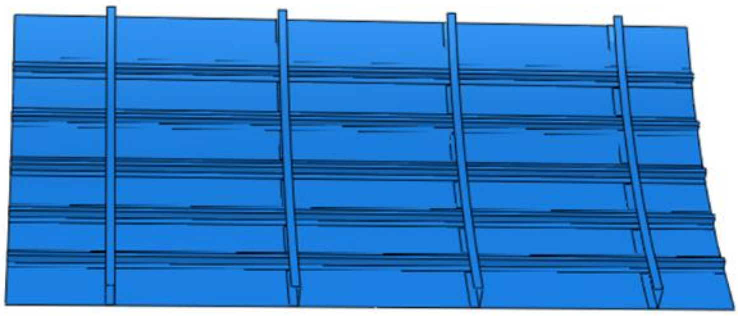


Figure 13. Panel Model

In order to evaluate the local stability behaviour of the optimized fuselage configuration, a Detailed Finite Element Model (DFEM) of the stiffened fuselage panel was developed. The DFEM panel model represents a local section of the fuselage skin reinforced by longitudinal stringers and circumferential frames. The boundary conditions applied to the panel model were designed to closely represent the edge conditions experienced by the panel within the global fuselage structure. These conditions were derived from the global fuselage model (GFEM) and simplified into cylindrical boundary conditions appropriate for panel-level analysis.

The applied boundary conditions are summarized below (Fig. 14):

- All three rotational degrees of freedom of the skin edges (both straight and curved edges) are suppressed.
- All radial displacements of the skin edges are constrained to represent the cylindrical shell behaviour of the fuselage structure.
- All three rotations of the frames and stringer end cross-sections located at the panel edges are suppressed.
- Tangential and axial displacements of one corner node of the panel are constrained in order to eliminate rigid body motion of the structure.

These boundary conditions ensure that the panel experiences deformation patterns that are consistent with those present in the global fuselage structure under the applied loading conditions. The section forces extracted from the HiFi fuselage barrel model were applied as distributed in-plane loads along the panel boundaries (Fig. 14). These loads represent the membrane force resultants acting on the fuselage skin due to the global structural loading.

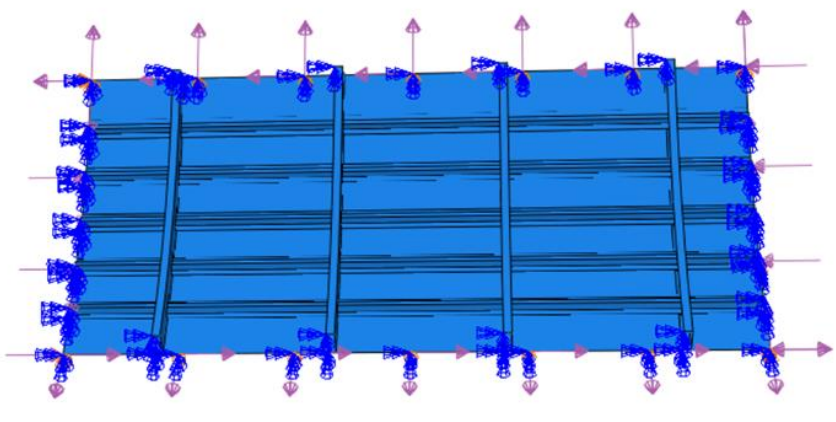


Figure 14. Panel Boundary Conditions

The panel geometry consists of multiple stiffened bays formed by the interaction between the fuselage skin, stringers, and frames. The same shell element formulation used in the HiFi model was adopted for the DFEM panel model to maintain modelling consistency.

The panel structure was discretized using shell elements with a refined mesh (Fig. 15 and Table12) in order to accurately capture local deformation patterns and buckling behaviour. The resulting mesh contains a sufficiently high element density to ensure reliable prediction of the structural instability modes.

Table 12. Panel Structure Parameters

Part	Mesh Size
Skin	10 mm
Stringer	7.5 mm
Frame	7.5 mm

Figure 15. Panel Structure with Refined Mesh

In the present study, the Lanczos eigenvalue solver available in Abaqus was used to compute the buckling modes of the DFEM panel model. The Lanczos solver is particularly well suited for large finite element systems because it efficiently computes a small number of eigenvalues and corresponding eigenvectors without requiring full matrix decomposition. Since the buckling analysis of large shell structures often involves a significant number of degrees of freedom, the Lanczos method provides improved computational efficiency and numerical stability compared with direct eigenvalue solution techniques. Consequently, it is commonly used in aerospace structural analyses involving buckling of large thin-walled structures [24]. In aircraft structural design, a commonly adopted stability requirement is that no skin buckling should occur below the limit load (1.0 LL). This requirement ensures that the fuselage structure maintains sufficient safety margin against instability during normal operational loading conditions. The eigenvalue buckling analysis performed on the DFEM panel model produced the results (Fig. 16) summarized in Table 13.

Table 13. Eigen Value for Buckling Analysis

Mode	Eigenvalue
Mode 1	1.5374

Acceleron Aerospace Journal || AAJ.11.2106-2613
Volume 6, Issue 3, pp (1765-1780)
E-ISSN- 2583-9942
https://dx.doi.org/10.61359/11.2106-2613

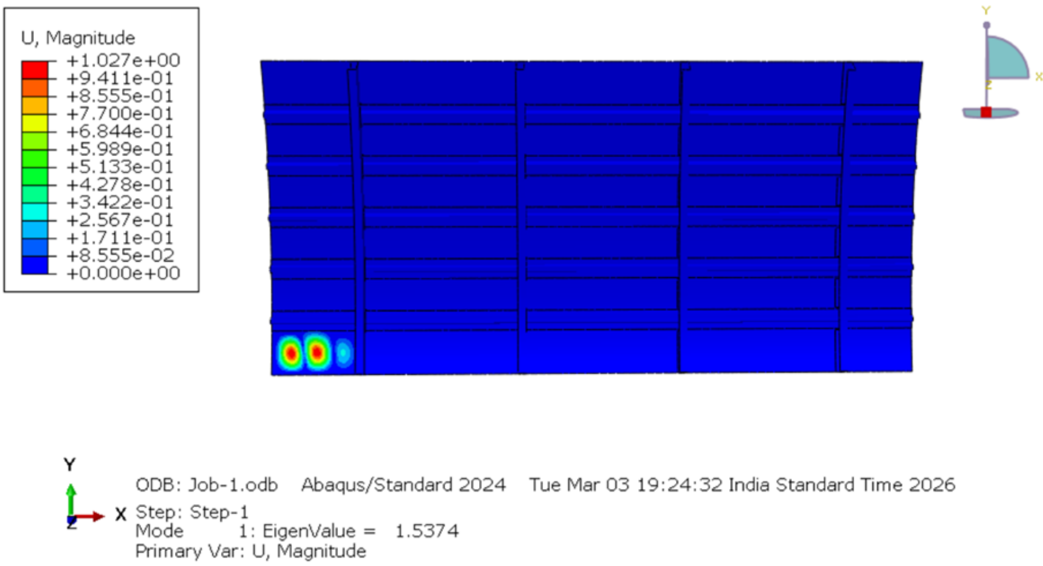


Figure 16. Eigen Value Buckling Analysis

The first buckling mode corresponds to the critical buckling condition of the fuselage skin panel. The eigenvalue of 1.5374 indicates that the applied loads must be increased by a factor of 1.5374 before the onset of structural instability occurs. The obtained buckling factor demonstrates that the optimized fuselage configuration satisfies the required structural stability criteria while maintaining reduced structural weight, thereby validating the effectiveness of the surrogate-assisted optimization framework.

8. Conclusion

This study presented a simulation-driven framework for the structural optimization of an aircraft fuselage structure using high-fidelity finite element modelling, surrogate modelling techniques, and global optimization algorithms. A detailed HiFi fuselage barrel model was developed using shell elements to represent the skin, stringers, and frames of the fuselage structure. Structural analysis was performed under the critical loading condition corresponding to Load Case 9, and the resulting simulation data were used to generate a dataset for surrogate modelling. A Sobol-based Design of Experiments strategy with 100 design samples was employed to efficiently explore the design space.

Surrogate models were developed to approximate the relationship between geometric design variables and structural responses. Polynomial regression provided accurate predictions of structural mass, while Box–Cox regression significantly improved the prediction accuracy for the nonlinear stress response. The surrogate models were then integrated into an optimization framework aimed at minimizing fuselage structural mass while satisfying stress constraints. Optimization was performed using both Particle Swarm Optimization (PSO) and brute force search methods. The results obtained from the two methods showed consistent optimal design regions, validating the reliability of the surrogate-assisted optimization approach. The optimized design was subsequently verified using the HiFi fuselage model, and section forces were extracted to construct a Detailed Finite Element Model of a representative stiffened panel.

The DFEM panel model, consisting of five stringers and four frames, was subjected to eigenvalue buckling analysis using the Lanczos solver. The analysis produced a buckling factor of 1.5374, confirming that the optimized configuration satisfies the structural stability requirement and does not experience skin buckling below the limit load. The results demonstrate that the proposed framework enables efficient exploration of fuselage structural design spaces while maintaining high modelling fidelity. By combining high-fidelity simulations with surrogate modelling and optimization techniques, the computational effort required for structural design studies can be significantly reduced without compromising the accuracy of the results. Future work may extend this framework to include additional structural constraints such as fatigue behaviour, damage tolerance, and multi-load-case optimization, enabling more comprehensive design studies for next-generation aircraft fuselage structures.

9. References

[1] Kasapoglou, C. (2010). Design and analysis of composite structures: With applications to aerospace structures. Wiley. https://doi.org/10.1002/9780470972700.
[2] Grihon, S., Krog, L., & Bassir, D. (2009). Numerical optimization applied to structure sizing at AIRBUS: A multi-step process. International Journal of Simulation and Multidisciplinary Design Optimization, 3, 432–442. https://doi.org/10.1051/ijsmdo/2009020.
[3] Haftka, R. T., & Gürdal, Z. (1992). Elements of structural optimisation. Springer. https://doi.org/10.1007/978-94-011-2550-5.
[4] Sobieszczanski, J., & Loendorf, D. (1972). A mixed optimisation method for automated design of fuselage structures. Journal of Aircraft, 9(12), 805–811. https://doi.org/10.2514/3.59080.
[5] Sobieszczanski-Sobieski, J., James, B., & Dovi, A. (1985). Structural optimization by multilevel decomposition. AIAA Journal, 23, 124–142. https://doi.org/10.2514/3.9165.
[6] Sobieszczanski-Sobieski, J., James, B., & Riley, M. (1987). Structural sizing by generalized multilevel optimization. AIAA Journal, 25, 139–145. https://doi.org/10.2514/3.9601.
[7] Beers, M., & Vanderplaats, G. (1987). A linearization method for multilevel optimization. Proceedings of the International Conference on Numerical Methods in Engineering.
[8] Barthelemy, J. F. M., & Sobieszczanski-Sobieski, J. (1983). Optimum sensitivity derivatives of objective functions in nonlinear programming. AIAA Technical Notes. https://doi.org/10.2514/3.8172.
[9] Sobieszczanski-Sobieski, J. (1988). A step from hierarchic to non-hierarchic systems. NASA-USAF Symposium on Multidisciplinary Analysis and Optimization.
[10] Sobieszczanski-Sobieski, J. (1990). Sensitivity of complex internally coupled systems. AIAA Journal, 28. https://doi.org/10.2514/3.10366.
[11] Hutchison, M. G., et al. (1994). Variable-complexity aerodynamic optimization of a high-speed civil transport wing. Journal of Aircraft. https://doi.org/10.2514/3.46481.
[12] Burgee, S., et al. (1996). A coarse-grained parallel variable-complexity multidisciplinary optimization paradigm. International Journal of High Performance Computing Applications. https://doi.org/10.1177/109434209601000402.
[13] Paiva, R. M., et al. (2010). Comparison of surrogate models in a multidisciplinary optimization framework for wing design. AIAA Journal. https://doi.org/10.2514/1.45790.
[14] Bettebghor, D. (2011). Optimisation biniveau de structures aeronautiques composites (Doctoral dissertation, University of Toulouse).
[15] Colson, B., et al. (2007). Composite panel optimization with nonlinear finite-element analysis. NAFEMS Seminar.
[16] Fleury, C., et al. (2010). Buckling optimization of composite stiffened panels. Proceedings of the 2nd International Conference on Engineering Optimization.
[17] Simpson, T. W., et al. (2001). Metamodels for computer-based engineering design: Survey and recommendations. Engineering with Computers. https://doi.org/10.1007/PL00007198.
[18] Bettebghor, D., & Bartoli, N. (2012). Approximation of the critical buckling factor for composite panels. Structural and Multidisciplinary Optimization. https://doi.org/10.1007/s00158-012-0784-6.
[19] MSC Software Corporation. (n.d.). MSC Nastran.
[20] Dassault Systèmes. (n.d.). SIMULIA Abaqus.
[21] Dassault Systèmes. (n.d.). Abaqus user manual.
[22] Bruhn, E. F. (1973). Analysis and design of flight vehicle structures. Tri-State Offset Company.
[23] Niu, M. C. Y. (1988). Airframe structural design: Practical design information and data on aircraft structures. Conmilit Press.

10. Conflict of Interest

The authors declare that there are no conflicts of interest regarding the publication of this paper.

11. Funding

This research was conducted independently and did not receive any external funding. All resources used in this study were internally available to the authors.