

Research Article

Metaheuristic-Based Topology Optimization of Fuselage Stiffeners using a Repairing Technique

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Abstract:

This study focuses on the topology optimization of aircraft fuselage structures during the preliminary design phase. The primary objective is to minimize structural weight while maintaining performance and safety standards. Success-History-based Adaptive Differential Evolution with Linear Population Size Reduction (L-SHADE) and Finite Element Analysis (FEA) are employed as the optimizer and structural solver, respectively. Multiple load cases, including landing, roll, turn, push-down, and pull-up, are considered to ensure the fuselage can handle all maneuver loadings. Displacements, stresses, and the buckling factor are incorporated as structural constraints. A topology repairing technique is developed to address non-continuous topologies and reduce infeasible solutions during the optimization process. The proposed methodology enhances the efficiency and robustness of topology optimization in the context of aircraft fuselage design, resulting in an average structural weight reduction of 18.99% from ten optimization runs, with the best run being 43.19% weight reduction.

Keywords: Fuselage optimization, Topology optimization, Finite element analysis, L-SHADE, Topology repairing, Frame structure

1. Introduction

The aviation industry is currently highly competitive, particularly in areas such as designing lightweight aircraft and optimizing efficiency. Each stage of the design process plays a critical role in meeting specific performance and safety requirements. The aircraft design process is typically divided into three key phases. In the conceptual design phase, the overall concept and feasibility of the aircraft are established by defining key parameters such as size, shape, weight, and general configuration. During the preliminary design phase, the focus shifts to optimizing structural details while ensuring safety in both structural and aeroelastic aspects. Finally, in the detailed design phase, more structural and assembly details are required to construct a prototype. Additional experiments, such as ground vibration and flight tests, will be performed once the prototype is complete to verify safety and performance.

During the preliminary design phase, the design of the aircraft fuselage is carried out [1–4]. By applying optimization techniques, a lightweight structure capable of withstanding maneuver loads. In today's competitive global market, companies are increasingly leveraging innovation to deliver cost-optimized products. Among these innovations, topology optimization stands out as a powerful method for achieving optimized conceptual designs. This approach enables material layout optimization, leading to weight reduction and cost savings.

As such, we will employ topology optimization in the design of the aircraft's fuselage [1–5]. The skeleton structure [2,5,6] is widely used in topology optimization due to its high performance-to-mass ratio. With optimal topology, the structure will be lighter while still capable of withstanding all specified load cases. One of the most crucial issues in

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topology optimization is the continuity of the structure. Since structural members can be removed during the optimization process, many designs become discontinuous or unstable. These designs are prone to failure or are regarded as infeasible during the optimization process. A previous study [7] developed a topology-repairing technique inspired by this issue. This technique repairs a high proportion of infeasible solutions, significantly simplifying the problem.

In topology optimization, the problems are mostly non-convex and difficult to solve using gradient-based optimizers. In such cases, metaheuristics [2,4–15], which are stochastic and more flexible, are employed. The selected algorithm for this study is L-SHADE [11], an extension of SHADE. L-SHADE is one of the winners of the IEEE Congress on Evolutionary Computation (CEC). Its variants [9,10,13,15] are later developed and has been proven to be a state-of-the-art single-objective solver.

From a review of the relevant literature, it was found that some recent studies have utilized topology optimization in the design of aircraft fuselage structures for analysis and development [16]. However, the studies are focused on designing a single part of a fuselage including thin-plate like 2D [3,4] and 3D [17,18] parts. Another study perform topology optimization of the rear part of the structure [5], but not the whole fuselage. There is a study [2] using a skeleton structure to design the whole fuselage structure but not utilizing topology optimization. The problem in the study only contains shape and sizing design variables. There is not any study utilizing topology optimization on designing the whole fuselage structure existed in the literature.

To address the research gap, our study focuses on the design of the entire aircraft fuselage structure through topology optimization, utilizing the L-SHADE algorithm, which is highly efficient and particularly suited for solving discontinuous and non-convex problems. This approach is integrated with structural analysis using Finite Element Analysis (FEA) with 3D frame elements.

Since structural members can be removed during optimization, many designs become discontinuous or unstable. These designs are prone to failure or are deemed infeasible during the optimization process. To overcome this challenge, a topology-repairing technique inspired by a study on pipe network optimization [7] has been developed. This technique repairs a high proportion of infeasible solutions, significantly simplifying the problem.

Another challenge when considering a large skeleton structure is the complexity of the problem. If sizing and topology variables are assigned separately, the number of design variables will double. With such a large problem dimension, optimizers may struggle to achieve global optima. To address this issue, a decoding method has been developed to convert a single set of design variables into both structural sizing and connection parameters. By merging sizing and topology variables into one set, the number of design variables is effectively halved, significantly reducing the problem's complexity.

The contributions of this paper are threefold:

- A design procedure for fuselage topology optimization is established.

- A topology-repair technique is developed to maintain the continuity and symmetry of the fuselage structure.

- A decoding method for design variables is proposed, reducing the number of variables—and, by extension, the problem's complexity.

The remainder of the article is organized as follows: Section 2 presents the related theory and problem formulation, Section 3 discusses the results, and Section 4 provides the conclusions.

2. Methodology

2.1. The Finite Element Analysis (FEA)

In this study, structural analysis is conducted using finite element analysis (FEA). Displacements, stresses, and buckling factors are determined through FEA. The governing equation for static analysis is expressed as:

$$Ku = f \quad (1)$$

In the equation above, $\mathbf{K}$ global stiffness matrix, $\mathbf{u}$ denotes the nodal displacement vector, and $\mathbf{f}$ represents the external force vector. Consequently, the nodal displacement can be determined as follows:

$$\mathbf{u} = \mathbf{K} \mathbf{f} \quad (2)$$

$$\mathbf{u} = \begin{bmatrix} \mathbf{K}_{11} & \mathbf{K}_{12} \\ \mathbf{K}_{12}^T & \mathbf{K}_{11} \end{bmatrix}^{-1} \mathbf{f} \quad (3)$$

The Matrices $[\mathbf{K}_{11}]$ and $[\mathbf{K}_{12}]$ can be expressed as follows:

$$\mathbf{K}_{11} = \begin{bmatrix} \frac{(A \times E)}{L} & 0 & 0 & 0 & 0 & 0 \\ 0 & \frac{(12 \times E \times I_{zz})}{L^3} & 0 & 0 & 0 & \frac{(6 \times E \times I_{zz})}{L^2} \\ 0 & 0 & \frac{(12 \times E \times I_{yy})}{L^3} & 0 & \frac{-(6 \times E \times I_{yy})}{L^2} & 0 \\ 0 & 0 & 0 & \frac{(G \times J)}{L} & 0 & 0 \\ 0 & 0 & \frac{-(6 \times E \times I_{yy})}{L^2} & 0 & \frac{(4 \times E \times I_{yy})}{L} & 0 \\ 0 & \frac{(6 \times E \times I_{zz})}{L^2} & 0 & 0 & 0 & \frac{(4 \times E \times I_{zz})}{L} \end{bmatrix}$$

$$\mathbf{K}_{12} = \begin{bmatrix} -\frac{(A \times E)}{L} & 0 & 0 & 0 & 0 & 0 \\ 0 & -\frac{(12 \times E \times I_{zz})}{L^3} & 0 & 0 & 0 & \frac{(6 \times E \times I_{zz})}{L^2} \\ 0 & 0 & -\frac{(12 \times E \times I_{yy})}{L^3} & 0 & -\frac{(6 \times E \times I_{yy})}{L^2} & 0 \\ 0 & 0 & 0 & -\frac{(G \times J)}{L} & 0 & 0 \\ 0 & 0 & \frac{(6 \times E \times I_{yy})}{L^2} & 0 & \frac{(2 \times E \times I_{yy})}{L} & 0 \\ 0 & \frac{-(6 \times E \times I_{zz})}{L^2} & 0 & 0 & 0 & \frac{(2 \times E \times I_{zz})}{L} \end{bmatrix}$$

where A is the cross-sectional area, E is Young's modulus, G is the shear modulus, J is the polar moment of inertia, L is length and I_{xx} , I_{yy} , and I_{zz} represent the area moments of inertia around the x-axis, y-axis, and z-axis, respectively, as calculated within the global coordinate system.

At this stage, once the displacement vector is determined, the stresses can be calculated as follows:

$$\boldsymbol{\sigma} = (\mathbf{D} \mathbf{B} \mathbf{T}) \mathbf{u} \quad (4)$$

where $\boldsymbol{\sigma}$ is a stress vector, $\mathbf{D}$ is an elasticity matrix, $\mathbf{B}$ is strain-displacement matrix, and $\mathbf{T}$ is transformation matrix. The stress vector will then be used to calculate the von Mises stress (σ_v) using the following equation:

$$\sigma_v = \sqrt{\frac{1}{2} [(\sigma_{xx} - \sigma_{yy})^2 + (\sigma_{yy} - \sigma_{zz})^2 + (\sigma_{zz} - \sigma_{xx})^2] + 3(\sigma_{xy}^2 + \sigma_{yz}^2 + \sigma_{zx}^2)} \quad (5)$$

All stresses in the right-hand side of (5) are stress component in $\{\boldsymbol{\sigma}\}$. In the analysis of deformation or buckling, buckling is a phenomenon that occurs when a structure or material is subjected to compressive forces or loads, causing it to suddenly bend or deform. Consider the following equation in the Local Coordinate System with a Fix-Fix boundary condition. The analysis is performed element-by-element to reflect the behavior within a fixed reference frame for each individual element.

$$(\mathbf{K} + \lambda \mathbf{G}) \mathbf{r} = 0 \quad (6)$$

where λ is the eigenvalue that represents the ratio of critical to applied loads, $\mathbf{G}$ is the geometrical stiffness matrix, and $\mathbf{r}$ is a buckling eigenvector.

The model used in this study is based on the General Atomics MQ-9 Reaper. The fuselage shape and aerodynamic model are shown in Fig. 1. The fixed boundary conditions of the finite element model are illustrated in Fig. 2. The fixed nodes are represented by red diamond markers. The members depicted with red lines are fixed and non-removable, while the black members are removable during the topology optimization process.

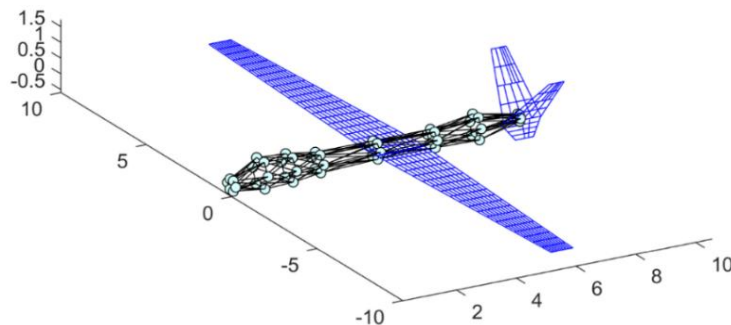


Fig. 1. MQ-9 Aircraft Model

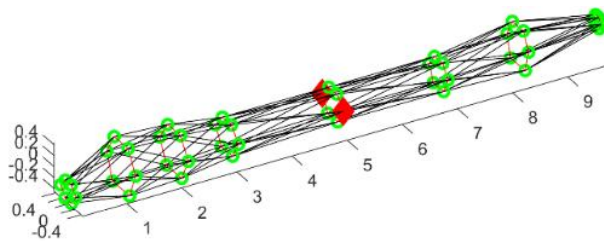


Fig. 2. The fixed nodes are represented by red diamond markers.

2.2 There are five load cases in this research, as follows:

Case 1: Landing involves the application of force at the bottom of the fuselage. The force is applied symmetrically on both sides, acting in the same direction along the z-axis, as shown in Fig. 3. Two nodes at the wing joint positions are fixed. The force applied during landing can be calculated using the following equation [19]:

$$F_v = \frac{W \cdot \frac{V}{g} \cdot k}{N} \quad (7)$$

where F_v is vertical force, W is the weight, V is the sink rate, k is the load factor, g is the acceleration due to gravity and N is the number of wheels and the force value is equal to 36,770 N.

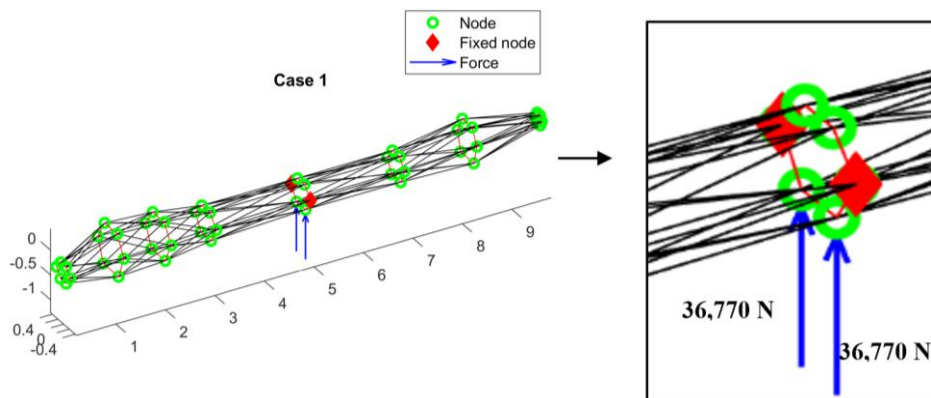


Fig. 3. Boundary conditions and loadings for Case 1: Landing

The aerodynamic maneuver loads of Case 2 to Case 5 are calculated with Vortex Lattice Method (VLM). Since the fuselage model is based-on General Atomics MQ-9 Reaper, the aerodynamic loads are calculate with the lifting and control surface size employed from MQ-9 data.

The forces calculated for Cases 2 to 5 are as follows: Case 2 is equal to 5689.5 N, Case 3 is equal to 993.4 N, Case 4 is equal to 6465 N and Case 5 is equal to 5689.5 N

Case 2: Rolling involves the application of force to the rear of the aircraft. The ailerons on both sides are adjusted by 20 degrees, and the calculated aerodynamic forces from the VLM are applied. These forces act in opposite directions along the z-axis, as shown in Fig. 4.

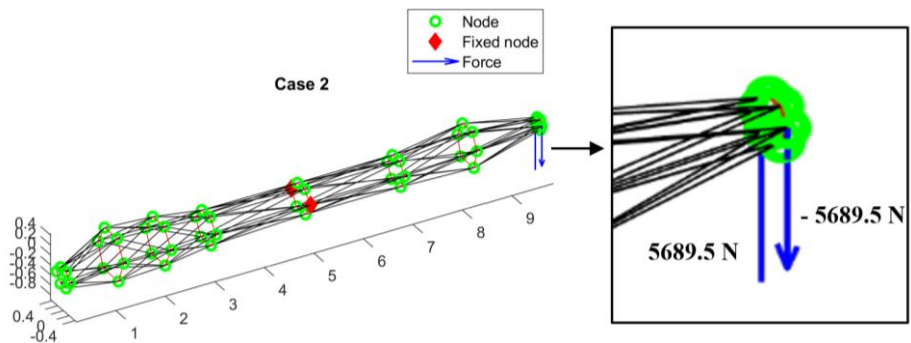


Fig. 4. Boundary conditions and loadings for Case 2: Roll

Case 3: Turning involves applying force to the rear of the aircraft. The rudder is adjusted to an angle of 10 degrees, and the force acts along the y-axis, as shown in Fig. 5.

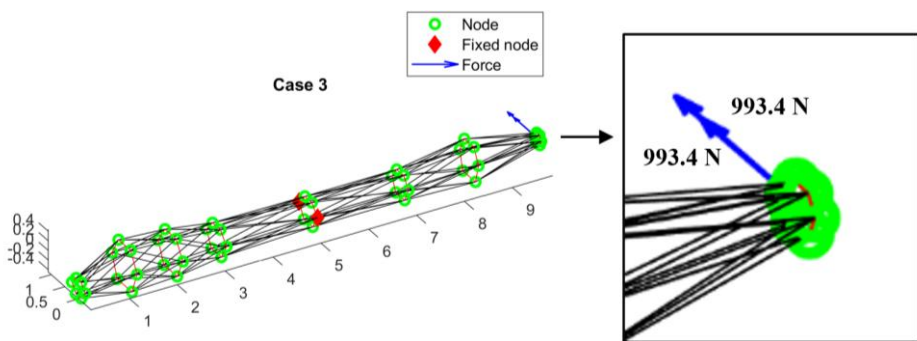


Fig. 5. Boundary conditions and loadings for Case 3: Turn

Case 4: Push-Down involves applying force to the rear of the aircraft. The elevators are adjusted to an angle of -20 degrees, with forces acting in an upward direction along the z-axis, as shown in Fig. 6.

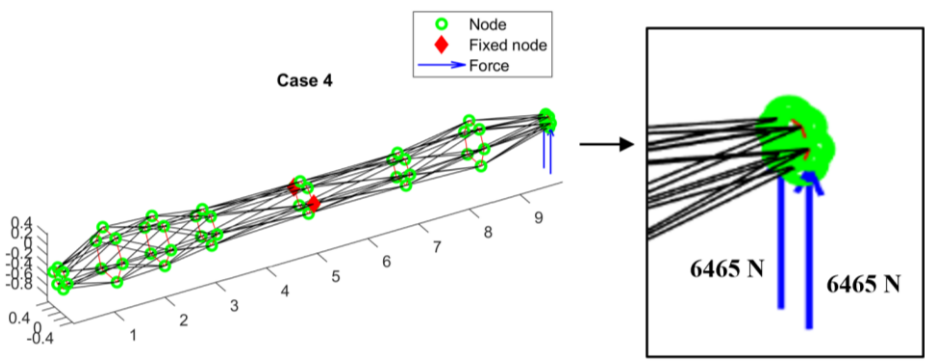


Fig. 6. Boundary conditions and loadings for Case 4: Push down

Case 5: Pull-Up involves the application of force to the tail of the aircraft. The elevators are adjusted to an angle of 20 degrees, and forces act in a downward direction along the z-axis, as shown in Fig. 7.

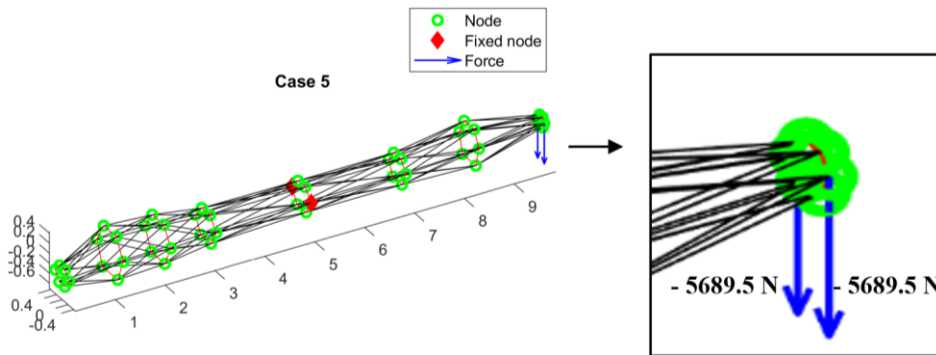


Fig. 7. Boundary conditions and loadings for Case 5: Pull up

2.3 Fuselage topology optimization problem

In this research, the aircraft's fuselage will be designed with the single objective of minimizing structural mass and the fuselage stiffeners are made of aluminum. The single-objective optimization problem addressed in this work can be written as:

Minimize $f(x)$ = the structural mass (kg).

Subject to:

$$g_1(x) = \sigma_{max} - \sigma_{allow} \leq 0$$

$$g_2(x) = \delta_{max} - \delta_{allow} \leq 0$$

$$g_3(x) = 1 - \lambda \leq 0$$

$$x \in \{150, 160, 170, \dots, 350\}$$

From the above equation, $g_1(x)$ represents Von Mises stress, σ_{max} and σ_{allow} are the maximum and allowable Von Mises stresses, respectively. For displacement $g_2(x)$, δ_{max} and δ_{allow} represent the maximum and allowable displacements, respectively. The allowable displacement, $\delta_{allow} = 0.1$ m, is specified to prevent excessive deformation, structural instability, or over-flexible structure. The buckling factor, $g_3(x)$ must be greater than zero to avoid global buckling related to bifurcation buckling theory.

The buckling Factor must greater than zero to avoid global buckling failure related to bifurcation buckling theory, so $g_3(x) = 1 - \lambda$ must be less than zero.

x is a vector of discrete design variables which are the diameters of the tube members while the lower (x_L) and upper (x_U) bounds are 150 mm and 350 mm respectively. Material properties including modulus of elasticity, allowable stress, Poisson's ratio, and density are 73.1 GPa, 440 MPa, 0.33, and 2780 kg/m³ respectively.

There are 174 members in the fuselage structure; however, considering symmetry in the x-z plane, the number of design variables is reduced to 71, as the detailed numerical results are provided in the Supplementary Data. The design variables correspond to the diameters of the tube members, each with a constant thickness of 5 mm. The fuselage design is divided into 8 sections.

2.4 Exterior Penalty Function Technique

To handle the design constraints, the constrained optimization problem is transformed into an unconstrained problem using a penalty function. The fitness function (f_p) is calculated using the exterior penalty function technique as follows:

$$f_p = f + C * (\max(\mathbf{g})) \quad (8)$$

where C is the penalty coefficient and $\max(\mathbf{g})$ is the maximum value of the constrained functions.

2.5 Optimizer

In this research, Discrete Optimization will be employed to identify the most optimal approach for organizing and selecting key components within the aircraft structure, subject to various constraints, in order to achieve the best possible results. The chosen algorithm is L-SHADE [11,13,15], which is one of the winners of the IEEE Congress on Evolutionary Computation (CEC) and this algorithm adaptive metaheuristic with a dynamic population size. The algorithm starts with a large initial population to maximize exploration. Then, the population size is linearly reduced to continuously enhance the exploitation ability. After the initial solutions are randomly generated, a new population is iteratively reproduced. Each individual in the population represents a solution that will be decoded into a fuselage design. Structural analysis via FEA is performed during the function evaluation process.

In the selection process, after all individuals in the new population are evaluated, the fitness of each individual is compared to that of the previous population. The better individuals will survive to the next iteration. At the end of each iteration, the adaptive parameters are adjusted based on the success history. The flowchart of L-SHADE is shown in Fig. 8. The calculations in the main loop will continue until the termination criterion is met. In this study, the criterion is the maximum number of function evaluations.

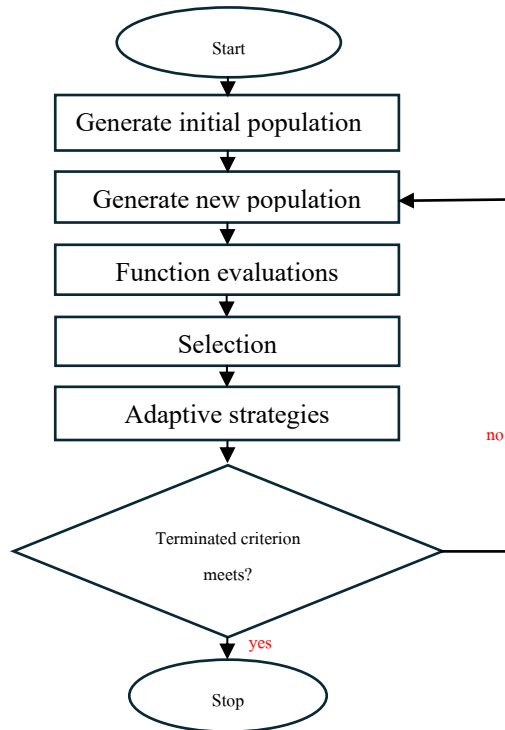


Fig. 8. L-SHADE flow chart

2.6 Function evaluation flowchart

First, material properties, boundary conditions, and loadings are pre-defined. The maneuver loads for all load cases are calculated using the VLM. It should be noted that the distributions for all load cases remain unchanged during optimization runs, as the shape and size of the lifting and control surfaces are not design variables in this study. The flowchart for the function evaluation is shown in Fig. 9.

The design variable decoding technique is proposed in this paper to handle topology optimization problems. The design variable vector ($\mathbf{x}$) contains design variables ranging from [0,1]. To address discrete topology and sizing optimization, these variables are converted into cross-section indices. The diameter ranges from 150 mm and 350 mm, with 0 mm included in the list for topology optimization purposes. Thus, the list of diameters is {0,150,160,170, ...,350}. The conversion is performed as follows:

$$\mathbf{t}_d = [N_d \cdot \mathbf{x}] \quad (9)$$

where, $\mathbf{t}_d$ is a vector of diameter indices, N_d is the number members in the diameter list, and $[\cdot]$ is a numeric operator that ceils the input to its nearest integer. During the optimization process, the design variables are iteratively updated. If a design variable is rounded to one, that means this index corresponds to zero diameter. In this case, a member will become inactive or be removed from the ground structure. If discontinuities or asymmetry occur, the structure is repaired using the technique described in Section 2.7.

After a vector of design variables is generated by L-SHADE, the FEA model is constructed. If the generated structure is discontinuous, it will be repaired using the topology repair technique described in Section 2.7. Following the repair process, the mass, displacements, stresses, and buckling factors for all load cases are computed with FEA. Finally, the fitness function (f_p) of a solution is calculated using Eq. (8).

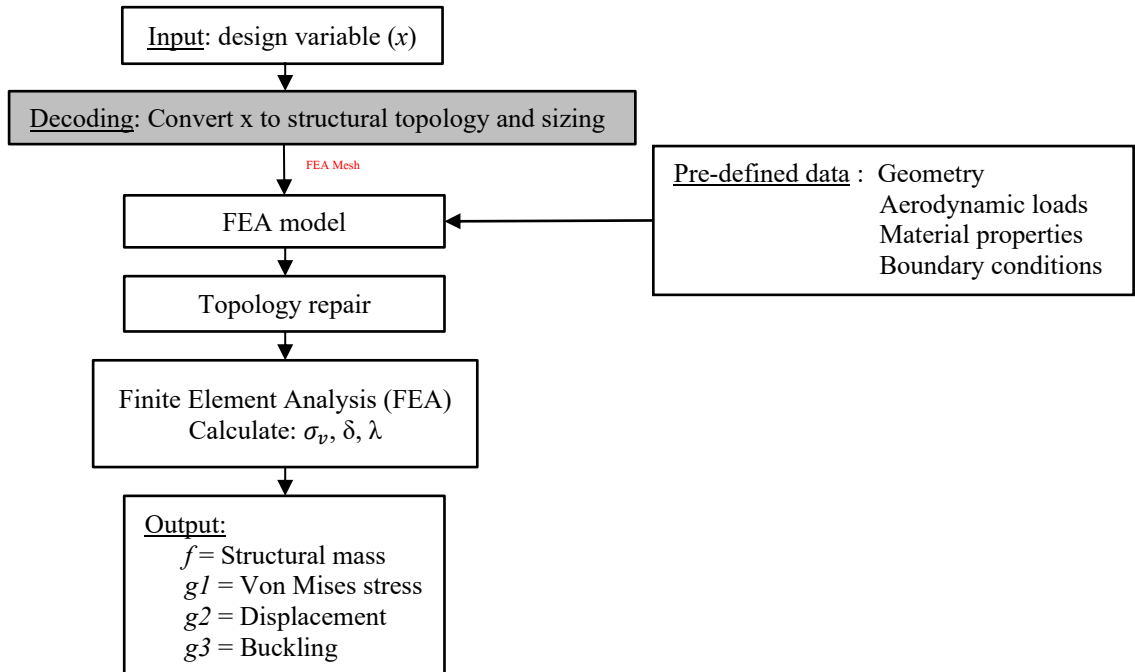


Fig. 9. Function evaluation flow chart

2.7 Topology repairing technique

The topology repairing technique developed in this study is inspired by [7]. However, since the original problem focuses on 2D pipe network optimization, which is significantly different from the fuselage optimization problem presented here, several improvements have been made to adapt the technique. The improvements are as follows:

- The code has been upgraded to handle 3D problems.
- Design variables encoding/decoding technique are developed.
- A symmetry detection and repair feature has been added. Symmetric members are automatically detected and repaired, ensuring the fuselage remains symmetrical.

The diameter of the members is assigned as the design variable in this study. The encoding/decoding technique is developed to address the topology optimization problem. During the optimization process, if a design variable (the member diameter) is within a specified threshold from the lower bound, the member is removed from the structure.

This approach eliminates the need for two separate sets of sizing and topology design variables, requiring only one set of variables. With this technique, the complexity of the problem, in terms of its dimension, is reduced by half.

From Fig. 10, the pre-repaired structure is shown on the left. The fuselage structure consists of 8 hexagonal ring sections. In the first step, members on all rings are repaired, as shown in the middle figure. Then, the discontinuous and non-symmetrical members are repaired in the second step, as shown in the right figure. The members used to connect discontinuous parts are randomly selected from the available members in the ground structure, as shown in Fig. 2. It should be noted that all ring members will always remain, while other members can be removed during the topology optimization process. The repair process ensures that all sections are connected and maintains symmetry in the structure.

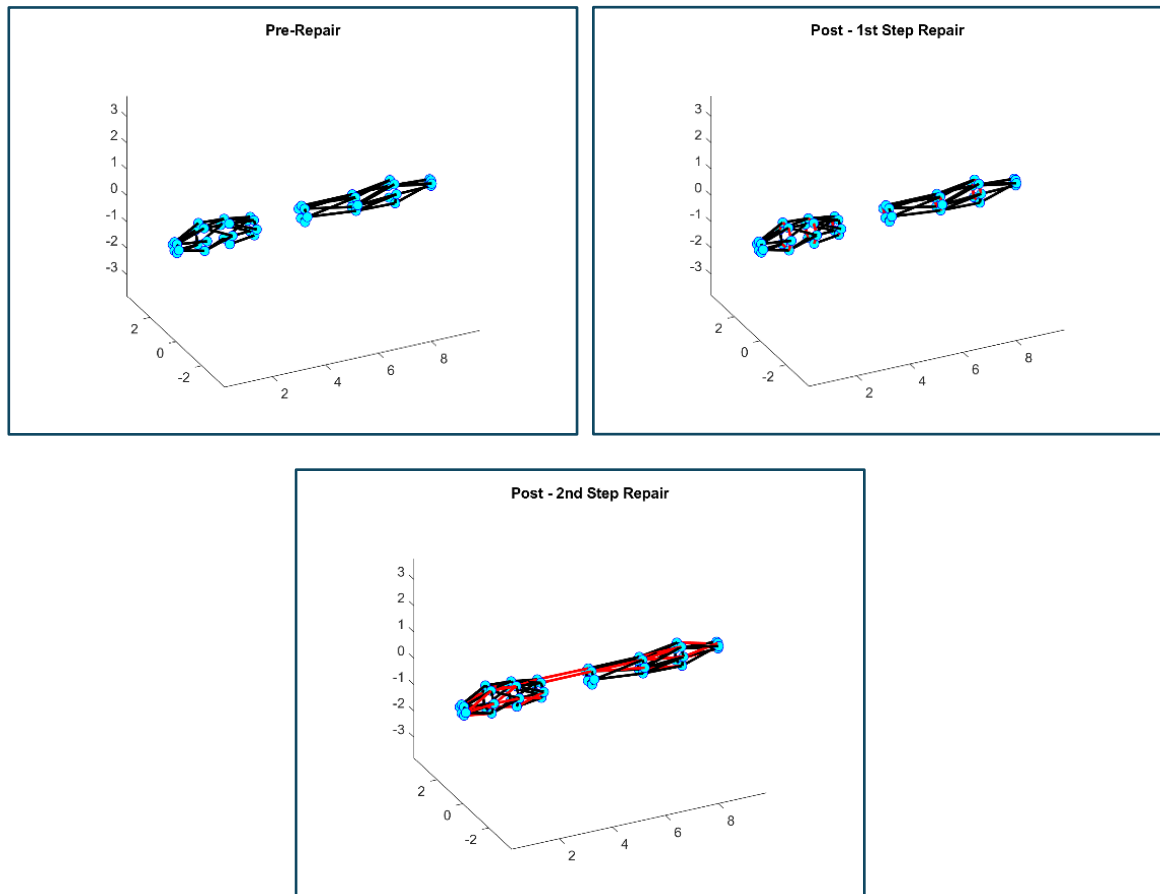


Fig. 10. Topology repairing process: pre-repair, 1st step repair, and 2nd step repair, respectively.

2.8 Experimental Setup

Ten independent optimization runs are performed using L-SHADE. The initial population size is set to 200, and the final population size is reduced to 4. The maximum number of function evaluations for all runs is set to 40,000 evaluations.

3. Results and Discussions

3.1 Optimization

The convergence of each L-SHADE run is displayed in Fig. 11. From the fitness history, it is evident that the algorithm is in its exploration phase during the first 10,000 evaluations because the convergence rate is rapid in this period. This can be observed by enlarging the image to see more clearly in Section A. In section B, after 10,000

evaluations, the convergence rate slows down, signaling the transition into the exploitation phase. This behavior is typical of L-SHADE, where the population size is initially large to enhance global search capabilities. As the optimization progresses and most of the search domain is explored, the population size gradually decreases to improve local search capabilities. However, results from L-SHADE still show some variation across different runs, likely due to the complexity of the topology optimization problem. Among the 10 runs, Run 2 achieves the lightest structure

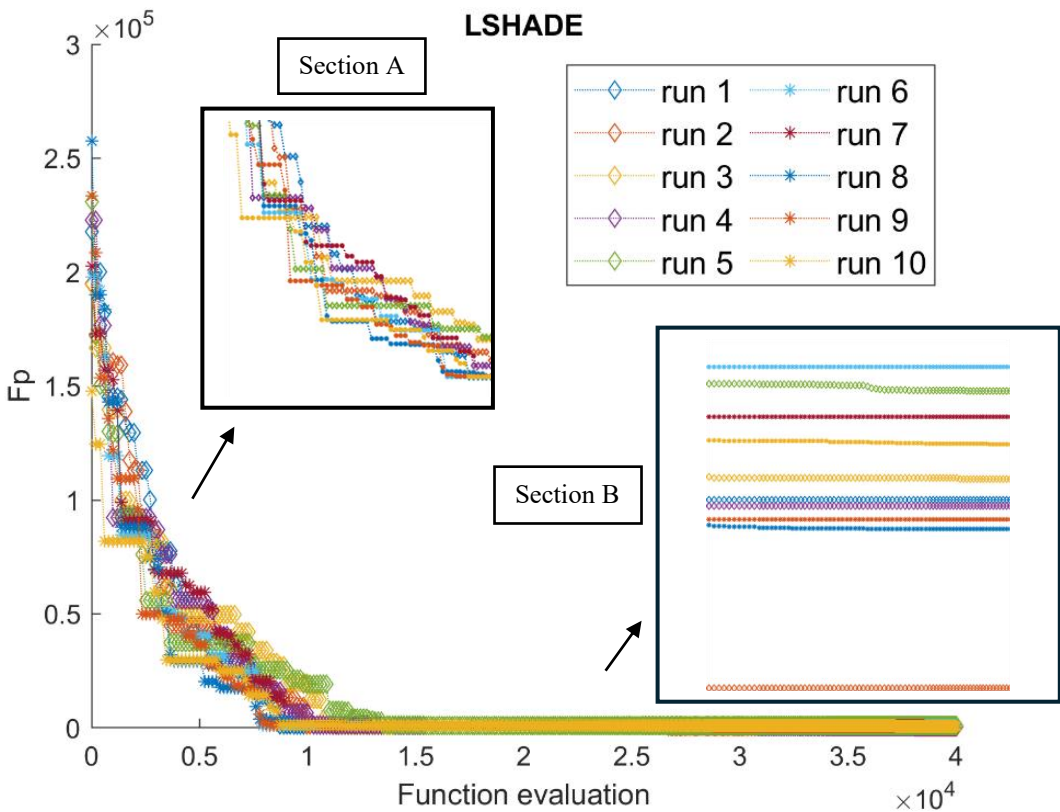


Fig. 11. The results of LSHADE for 10 iterations

Figure 12 compares the results of the two cases: Sizing Optimization only (blue line) and the proposed method, which combines Topology and Sizing Optimization (red line), to analyze the impact of incorporating topology optimization into the design process. In the initial phase, the blue line (Sizing Optimization) exhibits a more rapid decrease in fitness (F_p) compared to the red line. Although the first method has a faster initial convergence rate, it falls off in the second half of the optimization runs. In the later stages (as shown in the zoomed-in graph), the red line drops below the blue line. This suggests that the proposed method, simultaneous topology and sizing optimization which has more design diversity, outperforms the use of sizing optimization alone, yielding superior optimal results.

Table 1 presents the statistical results from 10 independent runs comparing the proposed method with the approach that uses only sizing variables. Both approaches consistently achieve 100% feasible solutions for all runs. The results align with the convergence plot, demonstrating that the proposed method outperforms the sizing design approach in terms of average, best, and worst results. With the proposed method, the average optimum mass is reduced by 18.99%, while the best optimum mass is reduced by 43.19%. The standard deviation results indicate that incorporating topology variables significantly expands the solution space, leading to a higher standard deviation in fitness values. The increased population diversity may cause the optimizer to exhibit greater variation or a higher standard deviation, but this trade-off improves the chances of finding a better optimum solution. Although the sizing-only method achieves a much lower standard deviation, the average mass is significantly heavier. For this reason, the downside of the proposed method can be considered negligible.

Table 2 presents a comparison of the best optimum results from both methods. All constraints, including maximum von Mises stress, maximum total displacement, and the buckling factor, are within allowable ranges. The most crucial

constraint in this study is the maximum von Mises stress. The maximum von Mises stress values for both methods are 439.39 MPa and 439.17 MPa, which are very close to the allowable limit of 440 MPa, while the other constraints remain well within their allowable values.

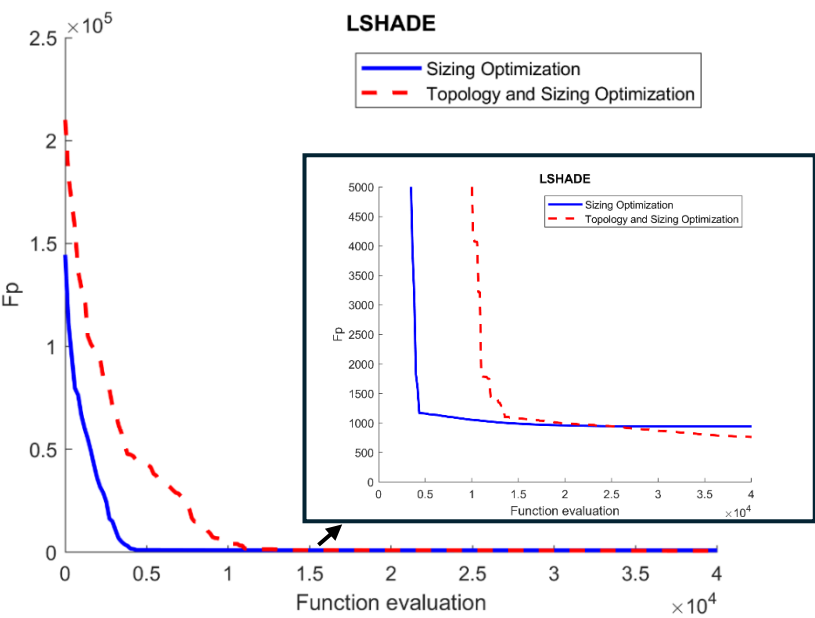


Fig. 12. Graph comparing cases with topology and sizing and only sizing

It is evident that the proposed method, which integrates both topology and sizing optimization, significantly reduces structural weight from 936.43 kg to 531.99 kg while maintaining acceptable levels of structural strength and safety. This demonstrates the method’s effectiveness in achieving a lighter and more efficient structure, making it well-suited for applications where high weight-performance efficiency is critical.

Table 1: Statistical results of fitness

	Average	Best (kg)	Worst (kg)	Standard deviation	% Feasible
Proposed Method (with topology and sizing variables)	764.0397	531.9870	897.0726	103.1425	100
Only sizing variables	943.2577	936.4285	952.4078	5.01335	100

Table 2: Comparison of Objectives and Constraints for Best Results from Both Methods

	$f = mass$ (kg)	$\sigma_{v,max}$ (MPa)	δ_{max} (m)	λ_{min}	Feasible
Proposed Method (with topology and sizing variables)	531.9870	439.39	0.0071	5.2319	Yes
Only sizing variables	936.4285	439.17	0.0056	9.6642	Yes

3.2 Von mises stress and deformed fuselage

In this section, two parts will be discussed as follows: Part 1 covers von Mises stress, and Part 2 focuses on the deformed fuselage.

Part 1, the von Mises stress results for the best optimum from the 2nd run are analyzed. The von Mises stresses are plotted on the deformed shape of the structure, with a scaling factor of 70 times applied to the deformation for better visualization. The von Mises stress serves as a failure criterion for ductile materials and is thus used as the failure criterion in this study. From Table 1, it is observed that the von Mises stress reaches its limit in all solutions, while the other constraints remain within their limits. This indicates that the von Mises stress is the most critical constraint in this study.

In Section C, as shown in Fig. 13, the maximum stress occurs at the wing root during landing due to the vertical forces applied during touchdown. This area experiences significant tensile and bending stresses as it transfers loads from the wings to the fuselage. To address these challenges, the wing root design was carefully optimized to minimize stress concentrations and enhance material strength. Reinforcements and optimized geometry were implemented to improve load distribution, ensuring the structure remains safe and functional under extreme conditions.

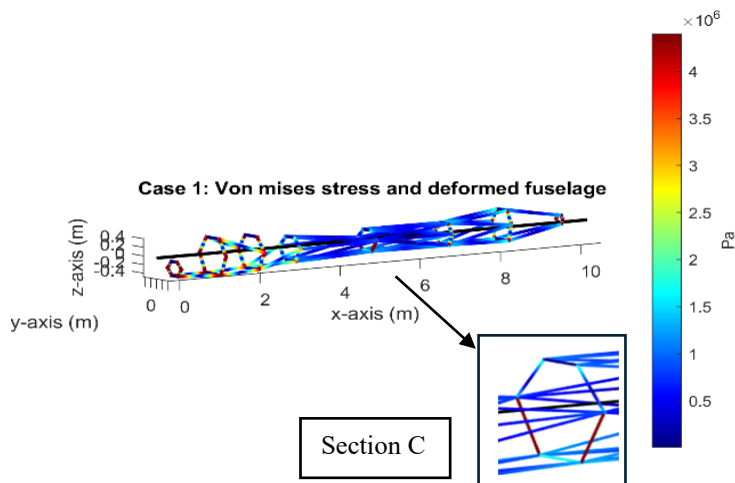


Fig. 13. Von mises stress and deformed fuselage of case 1: Landing

In Sections D and E, as shown in Figs. 14-17, the distribution of von Mises stress and deformation under different load cases is illustrated, highlighting the critical role of the tail section in load distribution. The optimization process ensures that torsional and compressive stresses are minimized by strategically reinforcing connections near the rudder and elevator, while maintaining symmetry through the topology repairing techniques. This approach reduces unnecessary mass in non-critical regions while preserving structural integrity under extreme forces.

Part 2 presents the results of the deformed fuselage, showing how the material changes shape under various applied forces in each case. In this section, we examine the images for irregularities in specific areas and assess whether they align with our predictions. If any areas exhibit excessive deformation, additional structural reinforcement may be required.

The deformation results illustrate the material deformation characteristics in each case. In Cases 1 to 3, the deformation follows similar patterns, primarily influenced by the weight of the aircraft structure. In contrast, Cases 4 and 5 show more pronounced deformation due to adjustments made to the model for better visibility of the results, combined with the forces acting on the rear section of the aircraft.

3.3 Optimum fuselage structure

From Fig. 18, the optimal fuselage structure illustrates a design that has been optimized to reduce weight while maintaining safety. In the figure, the structural links are represented by lines, with a noticeable concentration of darker lines in the central fuselage region. This concentration highlights the areas subjected to the highest forces due to the weight and load distribution within the structure.

Some lines are absent in the optimized structure as a result of the topology optimization process. These omissions indicate regions where material is not necessary to bear significant loads or support critical forces. By removing material from these non-critical areas, the overall weight of the structure is reduced without compromising its integrity or performance, ensuring the fuselage remains functional, safe, and efficient for aircraft use. It should be noted that the weight of equipment inside the aircraft is neglected in this study, which results in a lighter structure at the front part. In future studies, this additional weight should be considered to create a more realistic scenario.

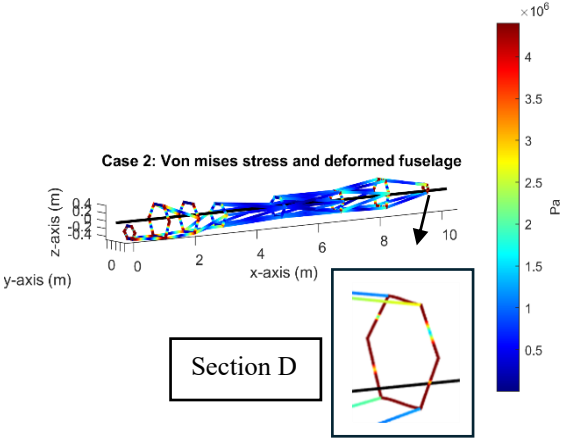


Fig. 14. Von mises stress and deformed fuselage of case 2: Roll

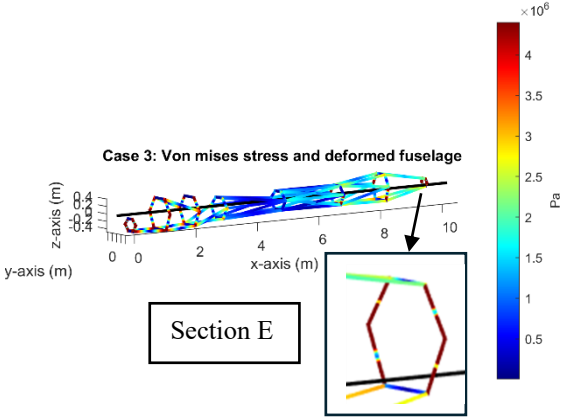


Fig.15 Von mises stress and deformed fuselage of case 3: Turn

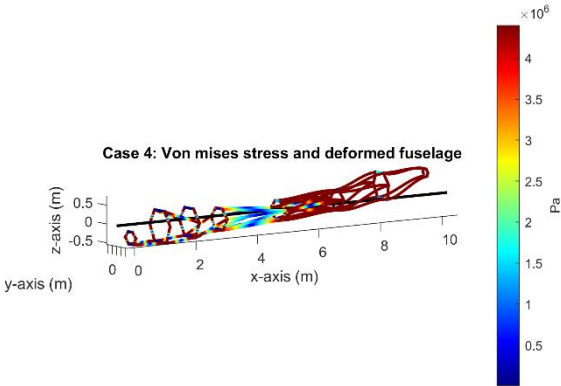


Fig. 16. Von mises stress and deformed fuselage of case 4: Push down

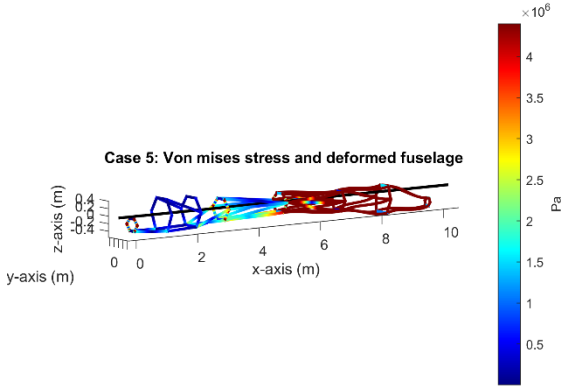


Fig. 17. Von mises stress and deformed fuselage of case 5: Pull up

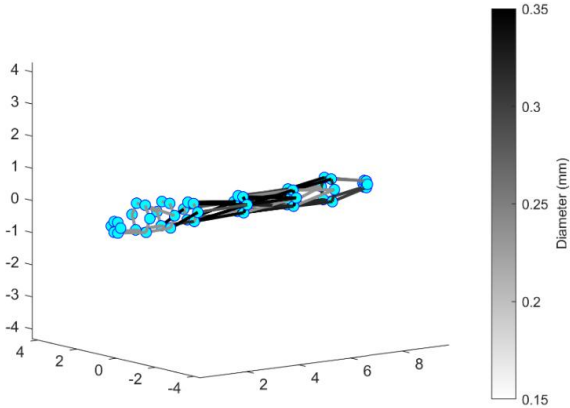


Fig. 18. Optimum fuselage structure

4. Conclusion

This research demonstrates the potential of topology optimization in the design of aircraft fuselage structures, significantly reducing structural weight while maintaining strength and stability. Based on 10 different optimization runs for each approach, the proposed method achieves, on average, 18.99% lower mass and 43.19% lower mass in the best run. For simplicity, in this study, the fuselage surface is not included in the FEA model. Additionally, aerodynamic loads are computed with deflected control surfaces using the Vortex Lattice Method (VLM). In future studies, a more realistic model incorporating fuselage wrapping surfaces, particularly those involving dynamic loads such as gust loads, will be necessary. These loads can directly impact the wing root area, where increased forces may be experienced, necessitating structural adjustments in these regions. Moreover, incorporating the weight of onboard equipment will improve the realism of the model. Additionally, developing models that account for uncertainties, such as material variations, is essential for achieving a structure that is both efficient and reliable under all operating conditions. Thus, this research provides a crucial foundation for the development of lightweight, cost-effective aircraft structures that meet the demands of the modern aviation industry. In the future, further improvements to the optimization algorithm may be needed, as experimental results indicate high standard deviation values, suggesting considerable fluctuation in the results and there will be an enhancement in the optimization process from a single-objective approach to a multi-objective approach.

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