

Article

Optimization-Driven Design of Composite Empennage Structures for Hybrid-Electric Aircraft

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Abstract

This paper presents the optimization-oriented structural design of aircraft empennages developed within the HERFUSE project, funded by the Clean Aviation Joint Undertaking. The study focuses on the horizontal and vertical tailplane of a hybrid-electric regional aircraft, considering the structural challenges introduced by distributed propulsion and novel integration requirements. The distributed propulsion layout influences the tail load envelope through powered-on maneuver/gust and asymmetric-thrust conditions, which contribute to the laminate redistribution in regions subjected to high bending–torsional demand. Starting from a conceptual structural configuration, the empennage is refined through a multi-objective optimization process. The structure is discretized into multiple regions, each characterized by independent laminate definitions selected from a predefined set of stacking sequences. The optimization aims to minimize the overall structural mass while ensuring adequate structural integrity under the project complete set of load cases. Structural performance is evaluated using a failure criterion suitable for composite materials, allowing the identification of critical regions and the redistribution of material accordingly. The results highlight the effectiveness of the adopted optimization strategy in improving structural efficiency, providing a refined configuration with a final mass of 433 kg, corresponding to an approximately 27.23% reduction with respect to the reference configuration coming from project specifications, while maintaining controlled failure index levels. The proposed approach demonstrates its suitability for supporting early-stage design decisions in next-generation hybrid-electric aircraft.

Keywords: empennage; structural optimization; composite structures; laminate design; hybrid-electric aircraft; multi-objective optimization; NSGA-II; Tsai–Wu criterion; weight reduction; aerospace structures

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1. Introduction

The transition towards sustainable aviation represents one of the major technological challenges of the coming decades. In particular, the need to significantly reduce

greenhouse gas emissions and fuel consumption has driven the development of novel aircraft configurations based on hybrid-electric propulsion systems [1,2]. These architectures are expected to play a key role in achieving the objectives defined by European initiatives such as Clean Aviation and Destination 2050, which target climate-neutral air transport within the next decades [3].

The introduction of hybrid-electric propulsion, including distributed propulsion systems and alternative energy carriers such as hydrogen, is fundamentally changing aircraft design. These technologies affect not only propulsion efficiency but also the overall structural layout, mass distribution, and load paths within the airframe [4]. As a consequence, traditional design approaches must be revisited, particularly for components such as the empennage, which plays a critical role in aircraft stability, control, and load balancing.

The structural design of empennages has been widely investigated in the context of conventional aircraft, with particular focus on lightweight composite structures and efficient load transfer mechanisms [5]. However, hybrid-electric configurations introduce additional complexities, including non-uniform and potentially asymmetric load conditions, as well as new integration constraints with the fuselage and propulsion systems.

In recent years, optimization techniques [6] have been increasingly adopted in aerospace structural design to improve performance and reduce weight. In particular, the use of composite materials enables a high degree of design flexibility through the definition of laminate stacking sequences and thickness distributions.

In the context of composite aerostructures, the use of optimization techniques is particularly relevant when dealing with components such as the empennage, where structural efficiency is strongly influenced by laminate configuration. Instead of adopting uniform or conservative sizing approaches, optimization enables a more targeted material distribution, allowing the structure to respond more effectively to the actual load environment. This aspect becomes even more critical in hybrid-electric aircraft configurations, where load distributions differ significantly from conventional layouts. In this scenario, the ability to locally adapt stiffness and strength through laminate design represents a key advantage in achieving lightweight and efficient structures.

Optimization techniques, such as the use of finite element analysis (FEA) and multi-disciplinary design optimization (MDO) [7], provide a framework to analyze the behavior of the laminate under different conditions.

Although optimization techniques have been extensively applied to composite aerospace structures, most studies available in the literature focus on local structural components or simplified laminate sizing problems. Limited attention has been devoted to the optimization of complete composite empennage assemblies at the preliminary design stage while considering many structural design regions and certification-driven loading conditions. Furthermore, hybrid-electric aircraft configurations introduce additional design constraints associated with modified aircraft architecture and load envelopes, requiring efficient material redistribution strategies to maintain structural performance while controlling weight.

To address these challenges, the present work investigates the optimization of a full composite empennage developed within the HERFUSE project. The proposed methodology combines finite element analyses and multi-objective optimization techniques to manage 110 structural design zones under 819 representative maneuver and gust load cases, enabling the identification of weight-efficient laminate configurations suitable for hybrid-electric regional aircraft applications.

The present work focuses on the optimization of the empennage structural configuration, starting from a conceptual baseline defined within the HERFUSE framework. A multi-

objective optimization approach is adopted to minimize structural mass while ensuring adequate structural integrity under representative loading conditions. The methodology is based on the definition of laminate configurations over multiple structural regions and the evaluation of structural performance using the Tsai–Wu failure criterion [8].

The paper is organized as follows: Section 2 describes the reference aircraft configuration, the development of the structural configuration, and the optimization baseline (motivation, design variables and constraints and optimization workflow); Section 3 presents the results and chosen optimized configuration; conclusions and future developments are finally discussed.

2. Materials and Methods

2.1. Reference Aircraft Configuration

The empennage design is based on the HERFUSE [9] Use Case B (UCB), which represents an advanced configuration for a hybrid-electric regional aircraft. The aircraft is designed to carry approximately 80 passengers and complies with CS-25 certification requirements.

A key characteristic of the UCB configuration is the adoption of a distributed hybrid-electric propulsion system, which significantly influences the aerodynamic and structural behavior of the aircraft. The presence of multiple propulsion units modifies the load distribution of the airframe and introduces additional constraints in terms of structural design and integration.

The empennage adopts a T-tail configuration, selected to reduce aerodynamic interference between the horizontal tailplane and the wake generated by the wing and propulsion system. This configuration also improves stability and control characteristics, particularly under off-design conditions.

The horizontal tailplane is conceived as a lightweight cantilever structure composed of a central load-carrying box with spars, ribs, and composite skins, designed to ensure efficient load transfer and a high stiffness-to-weight ratio. The vertical tailplane follows a similar structure, with particular attention to the interface region with the fuselage and the horizontal tailplane, where complex load transfer conditions occur.

This reference configuration (Figure 1) provides the baseline for the optimization process, enabling the evaluation and refinement of the structural layout under representative loading conditions.

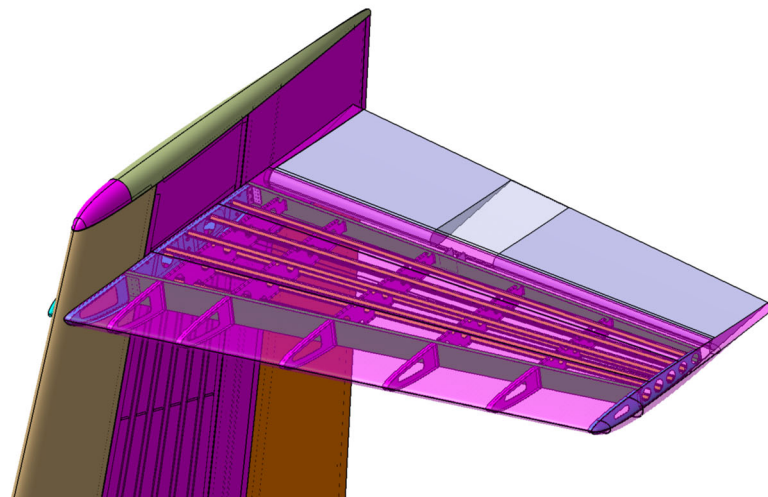


Figure 1. Reference CAD (software Catia V5).

The empennage structure is primarily based on carbon-fiber-reinforced polymer (CFRP) composites, selected for their high strength-to-weight ratio and suitability for aeronautical applications. Composite laminates are used for skins, ribs, and spars, allowing tailored stiffness through fiber orientation and enabling efficient load transfer under aerodynamic loading.

Manufacturing processes such as Liquid Resin Infusion (LRI) are considered for the horizontal tailplane, providing a good compromise between structural performance and production efficiency. To ensure the full reproducibility of the finite element simulations and to accurately feed the solver's composite failure criteria, a comprehensive characterization of the material system was implemented. The complete set of elastic constants, secondary engineering constants, and directional strength limits governing the carbon-fiber-reinforced polymer (CFRP) laminate is compiled and presented in Table 1.

Table 1. Properties of composite materials.

Composite Material	Unit	Carbon-Fiber-Reinforced Thermosetting Matrix
Ply thickness (th)	mm	0.202
Density (ρ)	ton/mm ³	1.52×10^{-9}
Longitudinal Young Modulus (E_1)	MPa	152,000
Transverse Young Modulus (E_2)	MPa	6900
Shear Modulus ($G_{12} = G_{31}$)	MPa	3300
Shear Modulus (G_{23})	MPa	1900
Longitudinal tensile strength (X_t)	MPa	2255
Longitudinal compressive strength (X_c)	MPa	667.9
Transverse tensile strength (Y_t)	MPa	17.3
Transverse compressive strength (Y_c)	MPa	83.1
Shear strength (S_c)	MPa	42.7

Metallic materials are adopted for structural joints and critical interfaces. Specifically, high-strength conventional aluminum alloys, namely Al 7075-T6, are exclusively used in localized regions and critical T-tail root interfaces subjected to high structural loads, where enhanced yield strength and stress-corrosion cracking resistance are required. To ensure a quantitative engineering framework, the baseline isotropic mechanical properties of this aerospace-grade alloy—including Young's modulus, Poisson's ratio, yield strength, and ultimate tensile strength—have been explicitly detailed in Table 2.

Table 2. Properties of Al 7075-T6 alloy.

Material Property	Unit	Al 7075-T6
Density (ρ)	ton/mm ³	2.79×10^{-9}
Young's Modulus (E)	MPa	70,000
Shear Modulus (G)	MPa	26,900
Poisson's Ratio		0.33
Yield Strength (σ_Y)	MPa	505
Ultimate Tensile Strength (σ_U)	MPa	570
Compressive Yield Strength (σ_{CY})	MPa	505

For the skin panels and their stringers, the adoption of a multi-rib co-curing manufacturing technology based on liquid resin infusion is proposed. This process enables the simultaneous curing of the skin and stiffening elements, resulting in a fully integrated composite structure.

The approach provides improved mechanical performance, particularly in terms of load transfer efficiency, structural continuity, and overall structural integrity.

From a manufacturing standpoint, co-curing reduces assembly time and minimizes the need for secondary bonding or fastening operations, leading to benefits in terms of overall weight reduction and increased production efficiency.

2.2. Development of the Reference Structural Configuration

The baseline structural configuration is defined through a simplified finite element representation, where skins, ribs, and spars are modeled using shell elements with equivalent orthotropic properties. The structural discretization, using Hypermesh [10], is consistent with the zoning adopted for the optimization process, enabling direct mapping between structural regions and design variables.

All tail components have been modeled on the *mid-surface*, according to the geometry of the individual parts.

During the modeling process, different finite element types were selected based on their specific capabilities and structural applications:

- Two-dimensional elements (Tria and Quad): These are used to model the rudder, front and rear spars, ribs, skin panels, longitudinal stringers, and fuselage attachment fittings, ensuring an accurate representation of the main structural surfaces.
- Rigid RBE2 elements: These are used to connect the vertical stabilizer to the fuselage via an SPC. They ensure rigid transfer of all six degrees of freedom (DOF), preventing any relative motion or spurious local deformations at the connection interface.
- Rigid RBE3 elements: These are used to distribute aeroelastic loads from the HERA [11] model onto rib and spar sections, with independent nodes exactly matching the load application points on ribs and spars (see Figure 2).

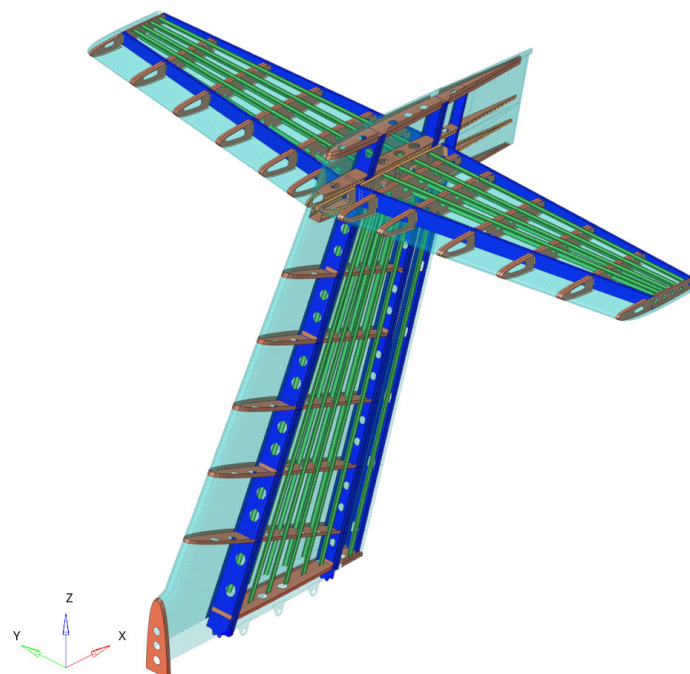


Figure 2. Finite element model overview.

To accurately represent the physical kinematics of the connection between the vertical tailplane and the fuselage, the SPC (Single Point Constraints) rather than using a full 6-DOF encastre, a selective constraint strategy was adopted: the lugs were constrained in DOFs 1, 3, 4, and 6 (X, Z translations and Rx, Rz rotations), while the front and rear spars

were constrained in DOF 2 (lateral Y translation). This approach prevents the occurrence of spurious local deformations and unrepresentative stress concentrations at the connection interface, ensuring a more realistic load path into the fuselage structure.

The structural loads employed for the analysis of the Vertical Tail Plane (VTP) are sourced directly from the dataset provided by Airbus Defence & Space within the framework of the HERA UCB (Clean Aviation) project. These values, which are fundamental for the preliminary sizing of the primary structure, constitute a comprehensive dataset comprising 819 distinct load cases. Specifically, flight loads were determined by integrating static maneuver analyses with dynamic simulations for Continuous Turbulence (CT) and Discrete Gusts (DTGs), ensuring full compliance with CS-25 certification requirements.

From an application perspective, the operational stresses are supplied as limit loads and are referenced to specific Monitoring Stations. For the tail assembly, equivalent external loads are applied by appropriately constraining the reference sections, thereby enabling the extraction of the reaction forces necessary for the structural calculation and subsequent sizing of the individual components.

The RBE3 elements were defined such that the dependent nodes exactly match those of the HERA project, both in terms of geometric position and nodal ID. This approach enabled the direct import of aeroelastic loads from the HERA model, ensuring full compatibility between the two models.

This methodology follows the same load application philosophy adopted for the single structural box (spar and ribs), providing a realistic load distribution across primary components without introducing numerical artifacts.

The integration between the Horizontal Tail Unit (HTU) and the Vertical Tail Unit (VTU) in the FEM model requires an accurate representation of interfaces and structural continuity of the tail assembly, achieved by modeling the connection regions between vertical and horizontal stabilizers using CBAR elements, which provide a simplified but effective representation of structural joints while ensuring proper load transfer and the compatibility of translational and rotational stiffness.

Considering the entire tail assembly, in regions of overlap between two or more components, an approach based on element duplication was adopted, with elements assigned to their respective structural components, in order to ensure a clear separation of individual contributions and a more realistic representation of the physical configuration of the overlaps; this methodology preserves the correct thickness of each component and improves the geometric and structural representation of the affected regions, enabling a more accurate characterization of local stiffness distributions and load paths.

The properties of the composite elements were defined using the PCOMP [10] property card, with material definition primarily based on a two-dimensional orthotropic material model (MAT8 [10]).

To guarantee the numerical accuracy and stability of the subsequent multi-objective optimization process, a rigorous element quality assessment was carried out across the discretized T-tail finite element domain. Given that elements failing to meet minimum geometric thresholds can introduce artificial stiffness or numerical singularities, specific shape parameters were continuously monitored and restricted within industry-standard aerospace targets. The final discretized model of the T-tail structure comprises a total of 64,783 nodes and 78,126 shell elements (predominantly Quad4 with minimal Tria3 transition elements), defining a representative mesh density for global structural behavior. The baseline mesh was developed using an average shell element size of 50 mm, a resolution chosen to capture localized stress gradients effectively without introducing an prohibitive computational overhead for the genetic algorithm. Elements with a length shorter than 5 mm were strictly avoided to prevent unnecessarily small time-steps or localized rigidities.

Regarding the mesh density verification, rather than executing a formal iterative mesh convergence study, the selection of the 50 mm baseline element size was strictly governed by established industry-standard aerospace guidelines for the global finite element modeling of transport aircraft components. Within this framework, numerical reliance and stress-field stability across critical root and junction regions are thoroughly secured by the strict enforcement of the rigorous geometric quality thresholds summarized in Table 3

Crucially, beyond the geometric grid filtering, a comprehensive Quality Assurance (QA) verification workflow was systematically executed to ensure absolute numerical compliance before initiating the optimization loop. The mathematical health of the assembly was secured by verifying that the Single Point Constraint (SPC) reaction forces perfectly balanced the applied global solver loads (OLOAD resultant), thereby eliminating any risk of unrepresentative internal load paths. Furthermore, a preliminary normal modes analysis (SOL 103) was implemented as a primary diagnostic tool to confirm correct part-to-part connectivity, validate relative component stiffnesses, and ensure the complete absence of unintended rigid body modes. The multi-stage criteria governing the geometric quality thresholds and the overarching FEM verification metrics are structured and detailed in Table 4.

Table 3. Mesh quality metrics and parameters.

Mesh Specs	Threshold
Mean Size	50 mm
Warpage	<5
Aspect Ratio	<5
Number of Tria	<1%
Skew	<60
Jacobian	>0.5
Min. length	>5 mm

Table 4. Systematic FEM verification protocol, solver execution checks, and structural modeling criteria.

Phase	Procedure/Check	Operational Target and Criteria
1. Pre-Processing	Mesh Connectivity	Shrink plot and coincident check to eliminate overlapping elements.
	Element Normal and Fiber	Alignment of shell normals and composite ply directions (<i>MCID</i>).
	Property Consistency	Validation of material cards (<i>E</i> , <i>G</i> , ν , ρ) and shell thicknesses in consistent FEA units.
2. Analysis Checks	Mass Validation	Model weight screening via <i>GRDPNT</i> card against CAD data.
	Numerical Conditioning	Matrix conditioning evaluation via Epsilon parameter ($\epsilon < 10^{-9}$).
	Load and Reaction Balance	Perfect balance between SPC reaction forces and applied loads (<i>OLOAD</i>).
	Dynamic Screening	Normal modes analysis (<i>SOL 103</i>) to ensure zero rigid body modes.
3. Post-Processing	Stress Field Continuity	Stress (Von Mises/Failure Index) and strain contour smoothing and peak gradient validation.

2.3. Optimization-Oriented Structural Refinement

The structural refinement of the empennage is driven by an optimization process aimed at improving the overall efficiency of the composite structure. Given the strong dependence of structural behavior on laminate stacking sequence, the optimization focuses on redistributing material across the structure according to the local load demand.

Each candidate configuration is evaluated through finite element analysis, where the failure index [12] is computed for all structural zones under a representative set of load cases. This allows the identification of critical regions and guides the iterative update of laminate configurations within the optimization loop. The process is repeated until convergence of the Pareto front is achieved.

2.3.1. Design Variable and Constraints

The tail structure is subdivided into 110 distinct zones, each representing a section of the tail that will be individually optimized. The optimization procedure assigns the most suitable laminate stacking sequence to each zone from a predefined set of 50 possible sequences. These sequences are based on previous research that has identified them as effective in similar applications. The goal is to determine the optimal combination of these sequences for each of the 110 zones, considering the specific stresses and strains each zone will experience during flight.

This optimization process is framed as a multi-objective problem, where the two primary objectives are to minimize the overall weight of the structure and ensure that it does not fail under the applied loads. By optimizing the laminate stacking sequence in each zone, the design can reduce material usage in low-stress regions while reinforcing areas subjected to higher stresses. Simultaneously, structural performance must be guaranteed by ensuring that no part of the tail fails, even under extreme flight conditions.

The structural weight is checked against a predefined upper bound, ensuring that the final design remains within acceptable limits for the tail's total weight. This constraint is critical because an overly heavy tail could negatively affect overall aircraft performance, including fuel efficiency, handling, and stability. To address the structural integrity requirement, the failure index for each zone is considered. The failure index quantifies the safety margin of the laminate design, based on the maximum stress the material can withstand before failure occurs. This index is typically calculated using a failure criterion—specifically, the Tsai–Wu criterion [8], which evaluates the failure in a composite laminate under varying loading conditions. In this optimization process, the second objective is to minimize the sum of the maximum failure indices across selected critical elements of the structure, alongside minimizing the total weight of the tail.

2.3.2. Optimization Workflow

The multi-objective problem is solved with the NSGA-II (Non-dominated Sorting Genetic Algorithm II) [13]. NSGA-II is a well-established evolutionary algorithm specifically designed for multi-objective problems. It works by generating a population of potential solutions (in this case, different laminate stacking sequences for each of the 110 zones) and iteratively evolving them to improve the design.

The algorithm sorts the population based on Pareto dominance, meaning that solutions are evaluated based on how well they balance the trade-offs between competing objectives. The two primary design goals are to minimize the structure's weight and the sum of the maximum failure index across the critical zones of the tail structure. The failure index [14] is a key metric for assessing the safety margin of the laminate, with higher indices indicating a higher likelihood of failure under stress.

The result of this optimization process is a Pareto front, which represents a set of non-dominated solutions where no other solution is better in both objectives simultaneously.

Each point on the Pareto front corresponds to a different trade-off between weight and failure index, allowing the most suitable design based on the specific needs of the aircraft.

Every point in the population represents a complete stacking sequence for all 110 zones of the tail structure. The population consists of 60 individuals, each encoding 110 stacking sequences.

To evolve the population, Simulated Binary Crossover (SBX) and Polynomial Mutation (PM) [15–18] are used as genetic operators. Since the stacking sequences are represented by integers between 1 and 50 (corresponding to the 50 predefined laminate stacking options for each zone), both operators are adapted to ensure that the resulting values remain integers.

Simulated Binary Crossover (SBX) operates on pairs of parent individuals, where each parent represents a full laminate configuration across all 110 zones. SBX generates offspring by combining genetic material from both parents. After performing SBX on the floating-point representations of the zones, the resulting offspring values are rounded to the nearest integer to ensure they correspond to valid stacking sequences.

Polynomial Mutation (PM) is then applied to the offspring. This operator introduces small random changes in the stacking sequences for each zone. The mutation is performed in the floating-point space, and the results are rounded to the nearest integer.

The final configuration is selected from this Pareto front to achieve the lowest possible structure weight while keeping the sum of the failure indices as low as possible, ensuring the structure's safety under operational conditions (a total of 819 Load Cases have been considered). The steps performed during the optimization analysis are highlighted in the next flowchart (Figure 3):

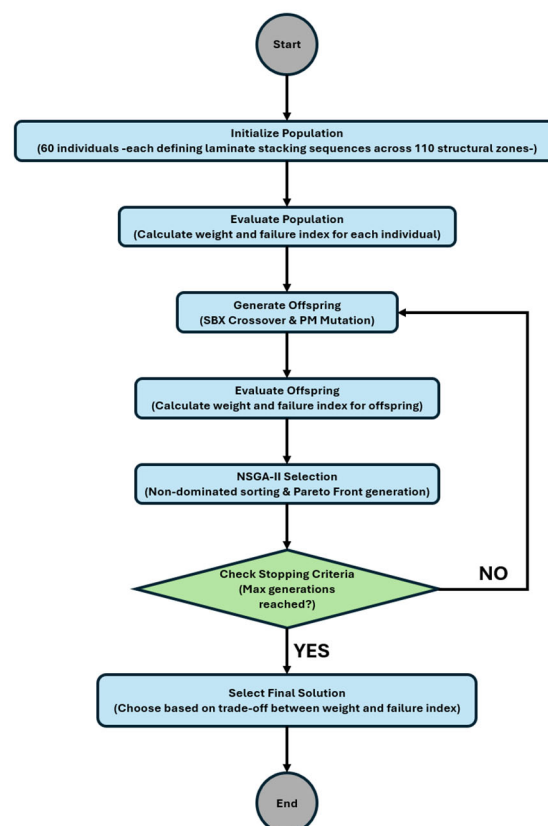


Figure 3. Optimization workflow.

The optimization process starts with the initialization of a population of 60 individuals. Each individual represents a full laminate stacking sequence across 110 zones (with values between 1 and 50). The population is evaluated through two parameters:

- Calculate Weight: Compute the total weight of the laminate structure based on the stacking sequences.
- Calculate Failure Index: Compute the sum of the maximum failure indices for the structure using failure criteria.

The workflow involves generating Offspring:

- Crossover (SBX): Perform Simulated Binary Crossover (SBX) between pairs of parent individuals. Offspring are generated by combining genetic material from the parents, ensuring offspring values are valid (integers between 1 and 50).
- Mutation (PM): Apply Polynomial Mutation (PM) to the offspring. Randomly perturb the stacking sequence values and round them to the nearest integer within the valid range (1 to 50).

Then, NSGA-II performs non-dominated sorting to select the best individuals based on Pareto dominance.

Criteria are verified with a check:

- If the maximum number of generations has been reached, proceed to subsequent step.
- If not, repeat the offspring generation process.

Finally, from the Pareto front, select the individual with the lowest weight while keeping the failure index as low as possible.

3. Results

Chosen Optimized Configuration

The following figures illustrate the results of optimization workflow. Figure 4 presents both the evolution of the offspring populations across generations, and the resulting final Pareto front composed of five layers, highlighting how progressively improved trade-offs between structural weight and failure index are achieved throughout the NSGA-II process. Within the same figure, the selected configuration is identified as a yellow star on the Pareto front, allowing a direct comparison between the chosen solution and the surrounding non-dominated alternatives.

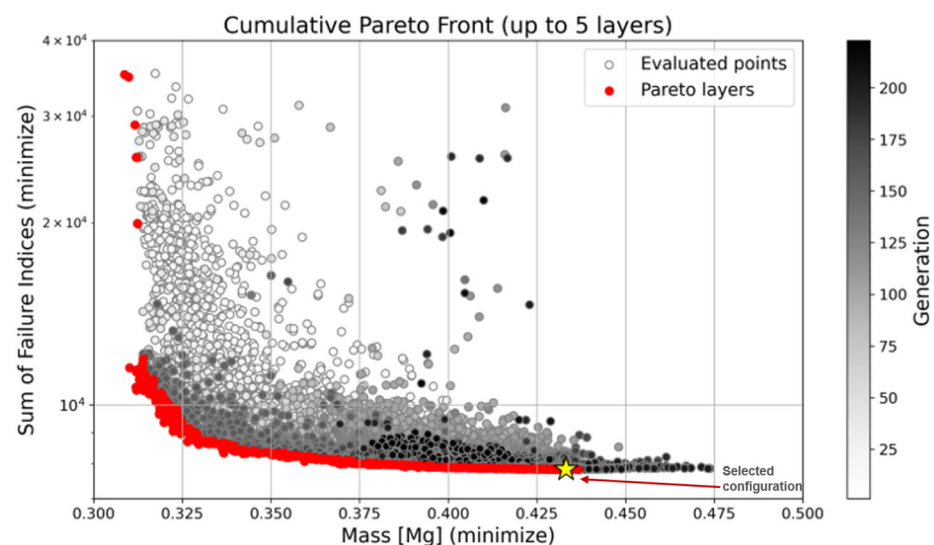


Figure 4. Multi-objective optimization history showing population trade-offs across generations and the distribution of non-dominated solutions.

Figure 5 focuses on the Failure Index distribution associated with the selected configuration, highlighting the structural regions where the response is most critical.

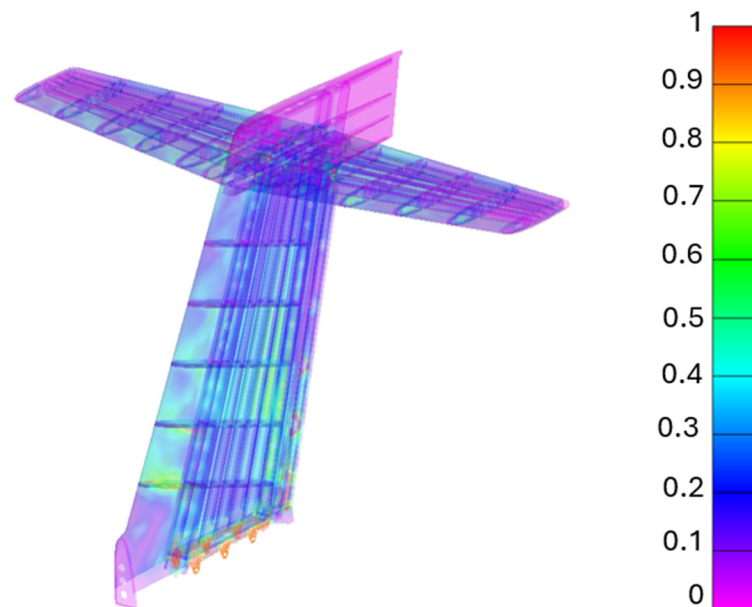


Figure 5. Failure Index distribution of the complete tail assembly under critical loading conditions.

Figure 6 provides a detailed zoomed view of the root region of the vertical tail, emphasizing local stress concentrations and laminate behavior in this highly loaded area.

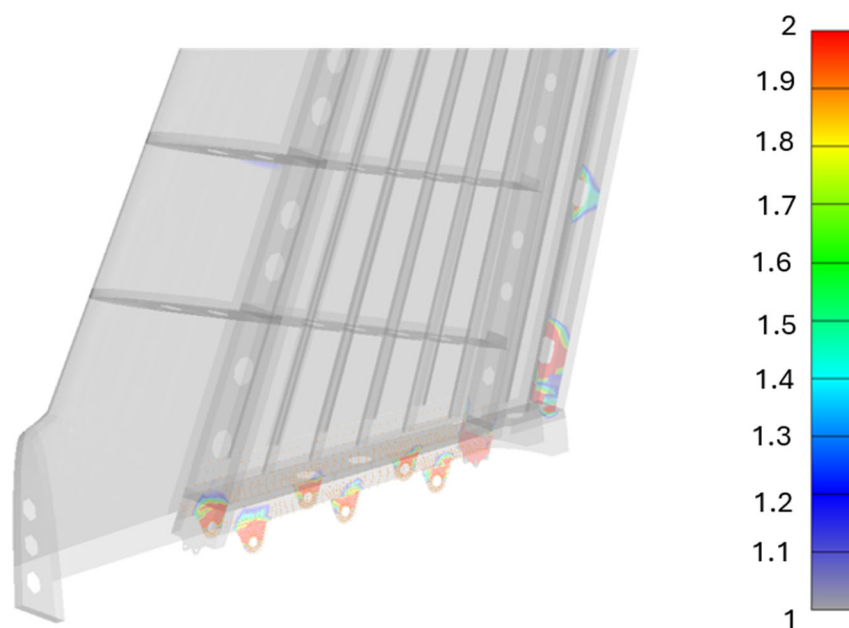


Figure 6. Detailed view of the Failure Index distribution in the vertical tail-fuselage attachment region under critical loading conditions.

Figure 7 similarly shows a magnified view of the intersection between the vertical and horizontal tail, illustrating the load transfer mechanisms and potential critical stress accumulation at this junction.

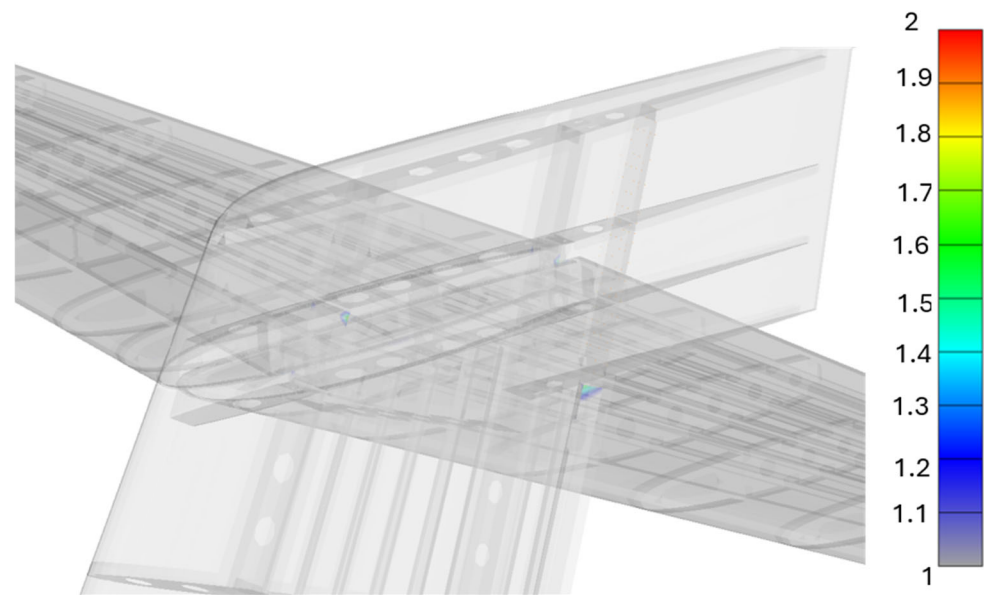


Figure 7. Failure Index distribution in the vertical–horizontal tail intersection region under critical loading conditions.

Finally, Figure 8 isolates and displays only those elements where the Failure Index exceeds unity, clearly identifying the failed or critical regions within the optimized structure. As shown in Figure 8, the regions exhibiting Failure Index values greater than unity are limited and localized, primarily at the horizontal tail plane–vertical tail plane and vertical tail plane–fuselage attachment interfaces. These critical areas can be effectively addressed through targeted local reinforcement and design refinements, thereby minimizing the impact on the overall structural weight. Together, these figures provide consistent progression from global optimization behavior to local structural assessment of the final selected design.

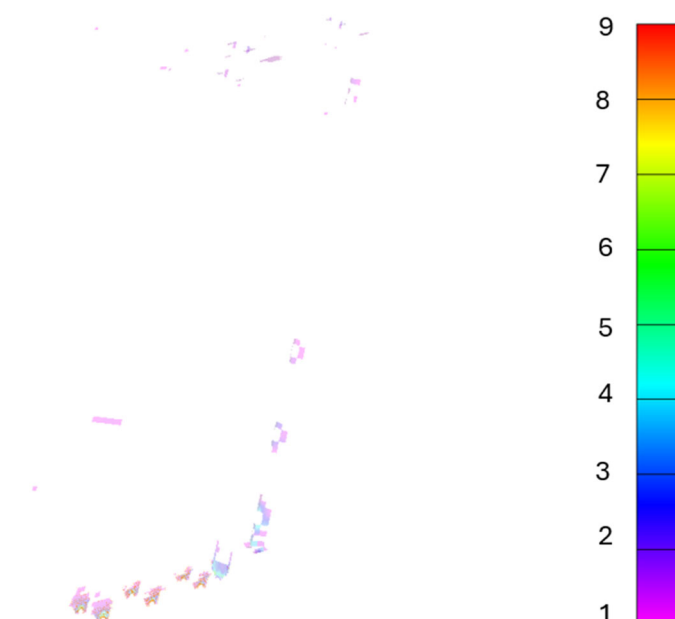


Figure 8. Elements with Failure Index greater than unity under critical loading conditions.

4. Discussion

The novelty of this work lies in the application of a multi-objective optimization framework to the structural design of a composite empennage, considering a high-resolution structural discretization and a large set of representative load cases. The proposed approach enables the efficient redistribution of laminate properties across the structure, providing improved weight–performance trade-offs compared to conventional design strategies. The multi-objective formulation leads to a set of non-dominated solutions defining a Pareto front, which represents the trade-off between structural weight and failure index [19].

The Pareto front highlights a clear trade-off between structural weight and failure index. As expected, lighter configurations tend to approach higher failure levels, while heavier solutions provide increased safety margins. This behavior confirms the consistency of the optimization process and reflects the underlying structural response of the composite layout.

The selected optimized configuration represents a balanced solution between these competing objectives. Compared to the baseline design, the optimized structure exhibits a more efficient distribution of material, with reinforcement concentrated in critical regions and reduced thickness in areas subjected to lower stress levels. This non-uniform distribution reflects the spatial variability of load conditions and confirms the effectiveness of the optimization approach in capturing the actual structural demand.

Critical regions are consistently identified in the vicinity of the vertical tail root and at the interface between the horizontal and vertical tailplanes. These areas are characterized by complex load transfer mechanisms and higher stress concentrations, in agreement with previous studies on composite empennage structures [20]. The optimization process effectively increases the structural capability of these regions, reducing peak failure indices and improving overall structural robustness.

From a broader perspective, the results confirm that optimization-driven approaches are particularly suitable for the design of composite aerostructures in hybrid-electric aircraft. The distributed electric propulsion layout modifies the tail load envelope by introducing powered-on maneuver/gust and asymmetric-thrust conditions. This affects the structural optimization process, since the critical load combinations may shift the optimum laminate distribution, especially considering the large design variability enabled by composite materials. These findings are consistent with recent developments in composite structural optimization, where laminate-level design variables are increasingly used to enhance performance [21–25].

The implications of this study extend beyond the specific empennage configuration considered. The adopted methodology can be applied to other airframe components and supports the integration of structural optimization within early-stage design processes. In particular, the ability to identify critical regions and optimize material distribution provides valuable guidance for subsequent detailed design phases and higher-fidelity analyses.

5. Conclusions

Nevertheless, some limitations must be acknowledged. The optimization is performed within a conceptual and preliminary design framework, and therefore relies on simplified modeling assumptions, particularly in terms of load representation and material behavior. Additionally, the use of predefined laminate configurations restricts the design space compared to fully continuous optimization approaches.

Future research will focus on the integration of the proposed methodology within multidisciplinary optimization (MDO) frameworks, including aeroelastic and aerodynamic constraints. Further developments will also address higher-fidelity composite modeling, detailed joint representation, and the extension of the approach to the full aircraft configuration.

Overall, the results demonstrate that the proposed optimization strategy provides an effective tool for improving the structural efficiency of empennage systems and supports the development of next-generation hybrid-electric aircraft.

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