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Wing Structure Optimization of a Truss-Braced Wing Regional Jet Aircraft with Strength, Stiffness, and Stability Requirements

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Abstract

A framework for high-fidelity wing structure analysis, sizing and optimization using the finite element method is developed. This framework, which can be applied to both cantilever (CLW) and truss-braced wings (TBW), takes into account the strength, stiffness and stability criteria. The strength sizing is based on shell and beam element properties optimization subjected to different load cases. For the stiffness sizing, a numerical methodology for converting the shell model to a beam model is developed and implemented. The stability sizing is based on engineering methods, and the strut section is sized to withstand compressive loads. The developed methodology is implemented for the sizing of the wing structure of CLW and TBW. Results show that the structural weight of a very high aspect ratio (aspect ratio of 20) truss-braced wing is higher than the high aspect ratio (aspect ratio of 11) cantilever wing by nearly 30% when sized for strength requirements. On the other hand, the flutter requirements have not affected the CLW design, while the outboard section of the TBW required more stiffness to pass this requirement.

1. Introduction

For several decades, the dominant design for successful commercial aircraft has been the "tube & wing" configuration. This configuration features engines mounted under the wings or at the rear of the fuselage. Over time, this configuration has undergone continuous refinement and optimization in order to maximize efficiency. However, reaching ambitious performance goals required for emission targets requires more drastic changes to the configuration. In recent years, various alternative configurations have been explored to identify the most promising configuration for future transport aircraft. These include the Blended Wing Body [1], Boundary Layer Ingestion [2], Turbo-Electric Propulsion [3], and Truss-Braced Wings (TBW) [4].

During the conceptual design phase, many configurations are investigated. This phase allows for considerable flexibility, even permitting changes to the overall architecture of the aircraft. To ensure cost-effectiveness while maintaining agility and adaptability, low-fidelity aerodynamics and structural analysis tools are typically employed at this stage. Additionally, the level of detail considered during this phase is limited. However, given the emerging requirements and the increasing costs associated with aircraft development, it has become crucial to develop tools and methods capable of designing and optimizing aircraft using high-fidelity approaches, even during the conceptual design phase. One of the aspects that should be addressed in the design of new aircraft, is the estimation of structural weight. This research focuses on developing a framework for high-fidelity structure sizing and weight estimation of wings of novel configurations.

The methods which are available for the calculation of weight can be categorized into three types [5]:

- Empirical Methods: These methods employ available aircraft weights breakdown and link them to the aircraft layout, material, and flight mission [6]. These equations are widely used for the design of conventional aircraft, where the database is rich.
- Analytical Methods: In this type of method, the structural loads are calculated by theoretical and engineering methods, and the wing structure at each station is sized for strength, stiffness, stability, and durability criteria [7].
- Numerical Methods: A finite element model, either beam elements or shell elements, is created, and the loads from each load case are applied, and the element thicknesses and cross-sections are sized based on those requirements [8].

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With current advances in computers and computation parallelization, more research is focused on the application of high-fidelity methods for the calculation of weight [9]. For this reason, a module should be available in the optimization process that can estimate the wing weight using Finite Element Analysis (FEA) accurately and fast enough, to make the optimization process feasible. In this process, a code dedicated to the calculations of weight is developed and integrated into the design framework.

The wing geometry, including the external surfaces and structure architecture, is extensively parametrized using knowledge-based engineering rules within a high-fidelity and industry-standard CAD tool. The application of parametric and associative modeling rules enables the possibility for fast geometry and structure architecture updates. This implementation covers both updating the existing geometrical elements (e.g., changing the wing spar location) and also the creation and deletion of structural elements (e.g., adding or removing ribs depending on the wing span).

The external wing loads are extracted by filtering from hundreds of load cases, from which the most critical load conditions are used for the sizing process. In the loading process, variations in weight, center of gravity, speed, altitude, and throttles level are considered. The Vortex Lattice Method is used for the estimation of aerodynamic load distribution along the wing span. In addition, the weight is distributed along the wing span, and discrete loads such as engine thrust and engine weights are also considered.

Based on the developed wing geometry and calculated loads, the wing finite element model using shell and beam elements is created automatically by Visual Basic scripts. This model is used in three sequential sizing processes: strength, stiffness, and stability. For strength sizing, the shell model and applied loads are used to find the optimum structural weight. The updated finite element model is then condensed (by applying geometrical, stiffness, and mass condensation) into the stick model, which is used in sizing according to flutter speed requirements. The results of the stiffness sizing are used to update the wing shell model, and the strut structure is sized for buckling requirements, particularly for negative load factors. The computed optimum wing is used along with the secondary items' weight to estimate the wing's structural weight.

2. Methodology

Overall Process In order to size the wing structure, a structure optimization framework is developed, which connects many external tools. The framework is based on the method developed by the authors, which is under review for publication [10], and the stability sizing is added in this research. At first, the wing's external surface is lofted using parametric modeling in the CAD module according to upstream wing planform design requirements. Based on the lofted surface, the internal structure architecture of the wing, which includes ribs and spars, is automatically modeled in CATIA using available automation tools. The created CAD model is then used in the structure module of the overall framework to create the finite element mesh. Based on the structure architecture, flight conditions and design requirements, the external load of the wing is estimated. These loads are used to size the wing's Principal Structural Elements (PSE) according to strength, stiffness, and stability requirements These sized properties are used to calculate the weight of the wing structure. The overall process for structure sizing is presented in Figure 1.

2.1 Framework Implementation

The optimization framework is implemented in the MATLAB environment. The developed framework, itself is a module within an aircraft optimization framework, and the details of this aircraft optimization are beyond the scope of this research. The framework input is a text file, and depending on the requested outputs, plots and results are stored in local files. The text-based approach reduces user interactions since all required information, inputs, and options are defined inside the input file before the execution of the code. For this reason, the file-based approach was selected over the interaction-based approach to reduce computational time and provide the capability for batch processing. The details of this process are mentioned in the following subsections.

2.2 Structure Architecture

A highly-parametrized CAD model of the aircraft's external surface and the wing structure architecture is developed in CATIA, to prepare the required data for the creation of the finite element model. The created CAD model is used to automatically update the aircraft geometry, based on the results sizing and design of the layout. The transfer of data to and from the CAD model is done through an integration framework in MATLAB. To this aim, MATLAB is connected to CATIA through the Microsoft Component Object Model (COM) architecture. The COM provides a framework for inter-process communication within the Microsoft Windows operating system. The data transferred through the COM connection is used to update the geometry and architecture automatically. This process is presented in more detail

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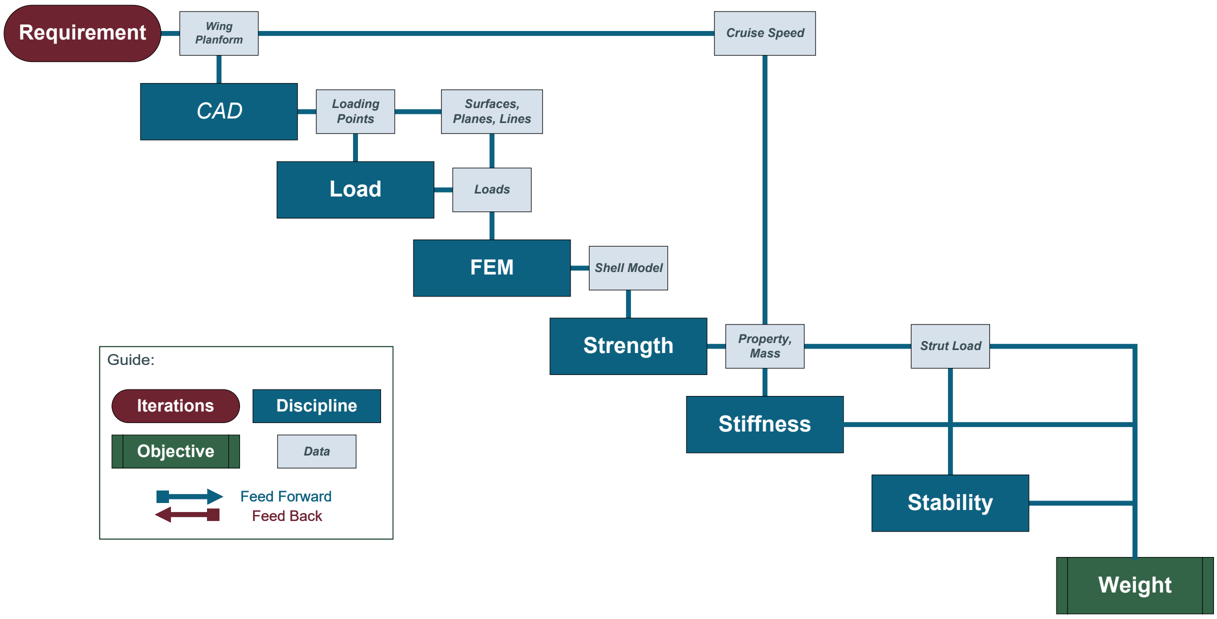
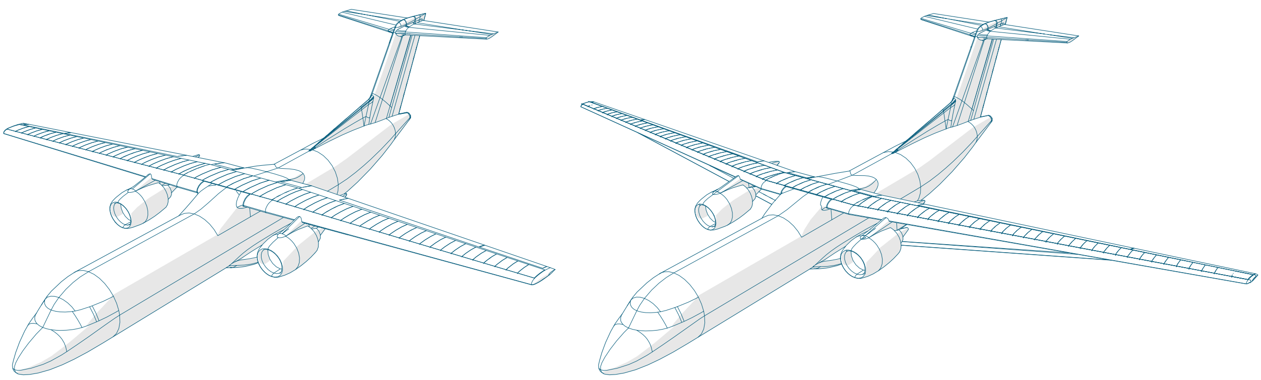


Figure 1: Structure Weight Estimation Flow Chart

in [11]. It is possible to generate different wing planforms, by variation of parameters, such as strut location, engine location, rib spacing, spars' location, and stringer spacing; two examples are presented in Figure 2.



(a) Cantilever Wing

(b) Truss-Braced Wing

Figure 2: Sample of Aircraft Parametrized CAD Model

2.3 Structure Loads

To size the wing structure for strength and stability requirements, the externally applied load to the wing should be known. This means that there should be a tool that makes use of existing tools (as much as possible) and calculates the loads associated with hundreds of load cases in a short time. To this aim, an automated wing loading process is developed. This code is implemented in MATLAB and interacts with existing and validated tools for aerodynamic trimming and load distribution. The reason that the existing tool is incorporated, is to increase the reliability of the code. This reliability comes with the cost and effort to develop and validate the interfaces with these tools. The developed code is capable of calculating hundreds of load cases in just a few minutes with conventional home computers.

The overall process for loading analysis includes the development of load cases based on flight envelope, load analysis, load distribution, load discretization, and load filtration. The developed process is capable of load analysis of novel truss-braced aircraft with novel wing configurations, where it is not possible to employ existing textbook methods. A short review of the loading process is presented here, and interested readers are referred to [12] for further

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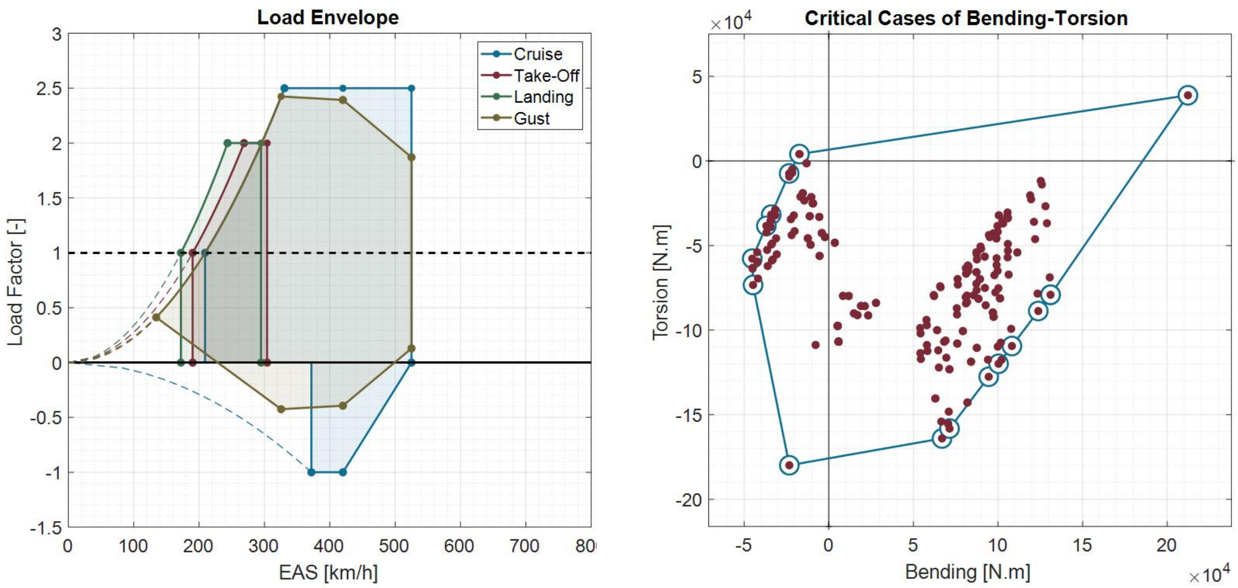
details.

Load Cases At the beginning of the load process, it is essential to define all possible load cases based on overall design requirements, i.e. cruise speed/Mach and dive speed/Mach, altitudes, etc. To this aim, clean maneuver, flap maneuver, and gust load envelopes are created for different weights and altitudes based on Part 25 requirements [13], see Figure 3a. Each corner point of these envelopes is added to the list of load cases considering variation in center of gravity, and throttle at each corner point. In addition to the envelope load cases, other types of load cases, such as roll, landing, and taxi loads are added to the list of load cases. In this research, the total number of load cases is 675.



(a) Load Envelope

(b) Truss-Braced Wing

Figure 3: Structure Load Process Results

Load Processing Software AVL [14] is used to calculate the trim conditions and flight states for each load case. This tool provides the capability of running a larger number of load cases within a short time. In addition to trim conditions, the AVL will provide the aerodynamic load distribution across the wing span, and this distribution later will be imported into MATLAB to calculate the aerodynamic load at each structural rib station.

Load Discretization In addition to the aerodynamic load that should be discretized over the span, the empty weight and fuel weight also should be discretized. Based on the aircraft weight configuration in each load case, the fuel weight in each wing segment (between each two ribs) is computed. The empty weight of the wing, which consists of structural and systems weight, is discretized along the sections. The empty weight, fuel weight, and aerodynamic load distribution are implemented using separate FORCE and MOMENT commands in the model. In other words, for each wing segment, and each one of empty weight, fuel weight, and aerodynamic load, there are separate FORCE and MOMENT commands.

Load Filtration To restrict the computational time, only those load cases that are critical for the wing sizing are considered. For this reason, a filtering process is implemented, to remove not-critical load cases from further analysis. To this aim, at each section, the combination of load components ($F_Z - M_Y$, $F_Z - M_X$, and $M_X - M_Y$) are plotted in a 2D plot. Using MATLAB commands, a convex hull is created surrounding these points, and all points on the boundary of this hull will be assumed critical, see Figure 3b. At the end of this process, the points load for each critical load case is exported to separate files, so the FEA modeler can read these files.

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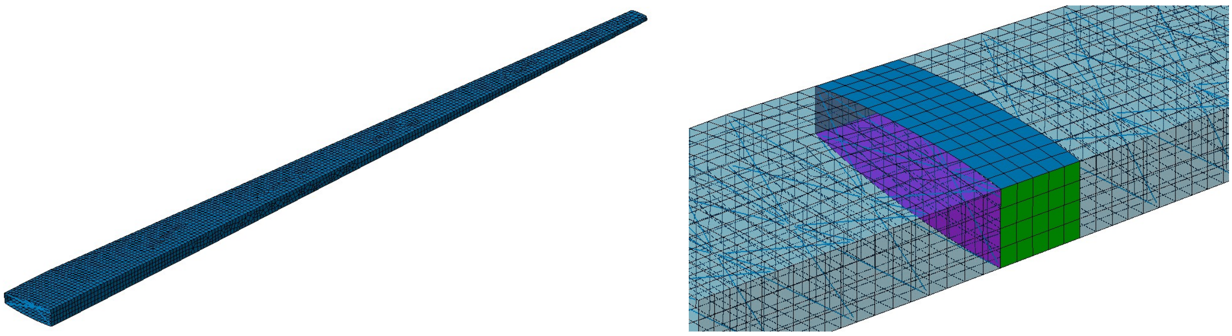
2.4 Structure Finite Element

The wing's main box structure consists of panels, spars, and ribs; and these structures should be idealized for the finite element model. The shell panels, which consist of skins and stringers, are idealized using smeared stringers [15] to reduce the number of variables and the computational time. The spar structure usually consists of webs, lateral caps, and vertical stiffeners. In the finite element model, spar webs are modeled using shell elements and the upper and lower caps are modeled using beam elements. The vertical stiffeners of the spars are assumed to be only at the intersection of spars with ribs, and the properties are included in the rib forward and aft cap. Similarly, shells are used for ribs webs, and beams are used for ribs' upper, lower, front, and aft caps. Internal rib stiffeners are not considered in this analysis, due to their local impact.

The wing geometry along with structure architecture needs to be converted to a finite element model automatically. To achieve this goal, a set of scripts in Visual Basic for Applications (VBA) language within LMS Virtual.Lab tool [16], which is a plug-in to CATIA, is developed. The whole process of creating the finite elements is commanded by MATLAB. The visual basic engine on LMS Virtual.Lab executes the creation of elements, loading points, interpolation elements, lumped masses, and restraints. The critical load cases, which are calculated in the previous step, are applied to the loading points of the finite element model. In the last step of this process, the Nastran input file in BDF format is generated. The results of this process are presented in Figure 4.



(a) Wing Box (b) Wing Rib

Figure 4: Finite Element Model

The properties of these created elements are the design variable for the sizing and optimization process. For the upper and lower panels, the thickness of the shell element, and the cross-section of each stringer are the design variables. These parameters are variable across the span, hence different properties are assumed between every two ribs. Similarly, the web thickness between each two ribs is a sizing property. To keep the computation time affordable, the sectional properties of spars caps, ribs caps, and ribs web are kept constant.

2.5 Sizing for Strength

The structure sizing is employed to calculate the minimum required material mass for a given wing planform, when subjected to the previously calculated external loads. The required load-carrying mass is calculated such that the maximum stress in all structural elements in all load cases is within the specified material stress limit. Nastran solution 200 (SOL 200) is used for the sizing process. The convergence of the weight parameter is presented in Figure 5a. The resulting design variables across rib stations are plotted in Figure 5b. The results of the strength sizing process are used to update the element properties for the next step.

2.6 Sizing for Stiffness

In this process, the wing structure is sized according to the flutter requirements. For this purpose, two modeling approaches can be used: the beam elements [17] or shell elements [18]. Shell elements will provide more accurate results, which come with higher computational and modeling time. On the other hand, the beam elements are more suitable for optimization frameworks, as they can provide acceptable results within an affordable time.

For employing beam modeling for the stiffness sizing, the shell model should be converted to an equivalent beam model, and this process is called condensation. To condense the shell box finite element model, a method consisting

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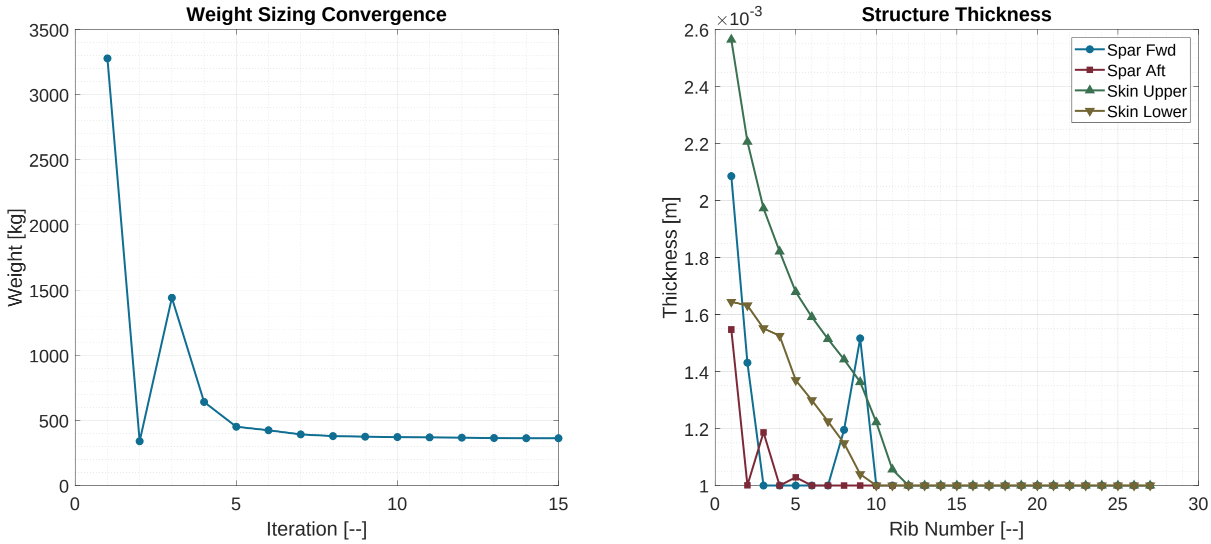

(a) Sizing Convergence (b) Properties Variation

Figure 5: Structure Sizing for Strength

of three sequential steps is employed: Geometry condensation, Stiffness condensation, and mass condensation. The overall process is presented in Figure 6, and will be discussed in the following paragraphs.

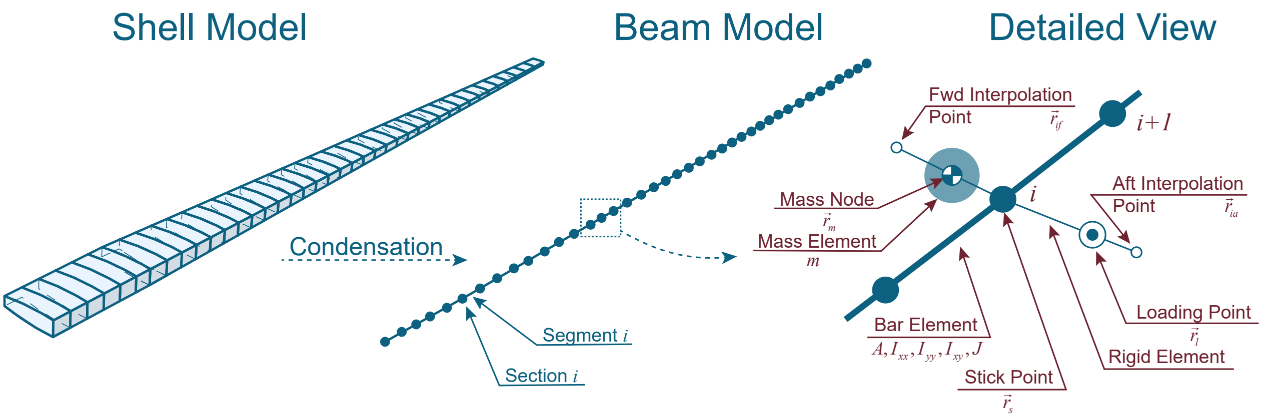

Figure 6: Condensation of shell finite element model to stick model

2.6.1 Geometrical Condensation

In the first step of the structure condensation, the elastic axis is calculated. This axis would be the location for the grid points of the condensed wing structure (stick points). In this research, a numerical method based on finite element analyses is implemented to calculate the longitudinal (chord-wise) location of the elastic axis at each rib station. The overall process for finding the elastic axis is presented in Figure 7. The lateral coordinates of the elastic axis are defined by the rib plane, and the vertical location is defined by the chord line. To find the longitudinal location, the coordinates of a point where an applied shear force will result in no sectional twist should be calculated.

To achieve this goal, six additional points, which are distributed equally along the chord, are created at each rib in the shell model, and these points are named geometrical condensation test points. These test points are connected with the rigid element to the rib loading point, while the loading point itself is connected to rib grids with the interpolation element (RBE3). For each test point, a load case is created, and in that load case, only that point is loaded with a vertical unit force. In this load case, the nodes located in the previous section are clamped. This means that 6 load cases are required for each rib, and the total load cases will be $6 \times n_{rib}$. The developed framework exports the solution

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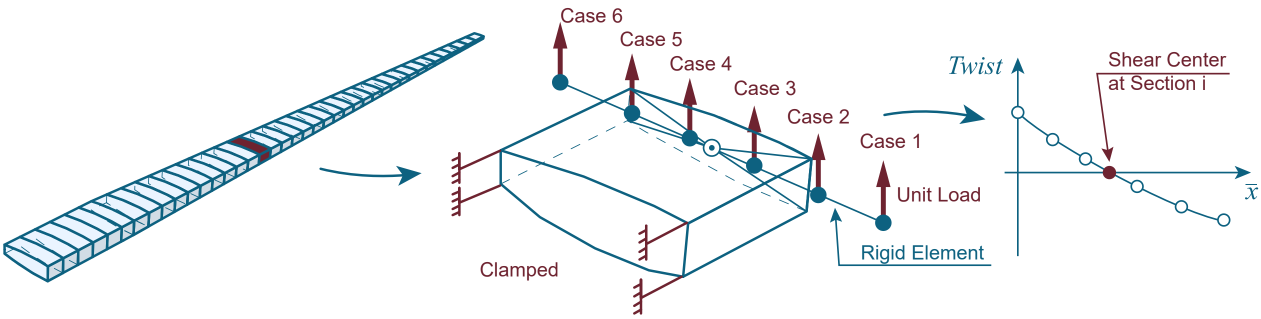
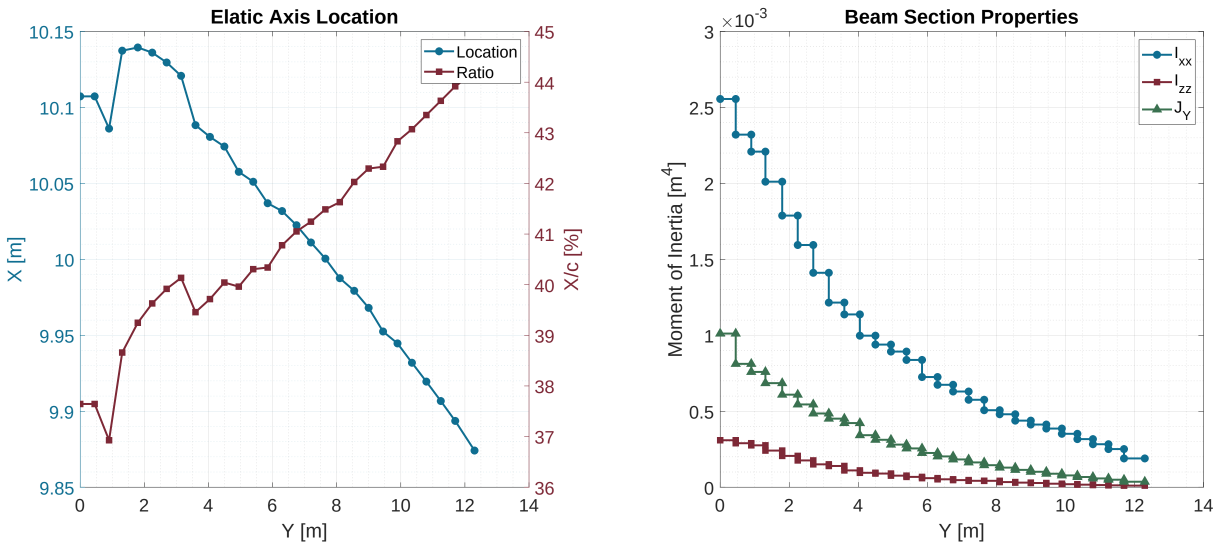


Figure 7: Geometry Condensation Process

in the BDF file, and the Nastran is called to calculate the deformations and displacements of the points. Based on the results of Nastran, the section twist due to each of these loads is calculated, and the point along the chord where the twist would be zero is found numerically using interpolation. The calculated result for a typical TBW and CLW configuration is presented in Figure 8a.



(a) Elastic Axis Location

(b) Beam Model

Figure 8: Geometry and Stiffness Condensation Results for Canti-Lever Wing

2.6.2 Stiffness Condensation

In the next step, the section properties of the beam model should be calculated to capture the static and dynamic characteristics of the original shell model. There are many methods available in the literature for stiffness condensation [19, 20]. In this research, the method presented in [21] is implemented. In this method, for the calculation of beam properties, a stiffness coordinate system is created for each section. The origin of this axis system is on the shear center, z_s direction is aligned with the elastic axis, x_s direction is toward the leading edge, and y_s completes the right-hand rule. Similar to the geometry condensation process, six load cases are created at each of the beam points. Three unit force and three unit moment loads are created in each of axes of the the defined coordinate system, and each one will be analyzed separately in a load case. In total, $n_{rib} \times 6$ load cases are required to condense the wing box. The process is presented in Figure 9. The Nastran is called to calculate the displacements, and the calculated deflections are then used to compute the section properties. The section properties, including I_{xx}, I_{yy}, I_{xy}, J, and A are calculated using the formulations provided in [21]. The results for stick properties for the cantilever wing are presented in Figure 8b.

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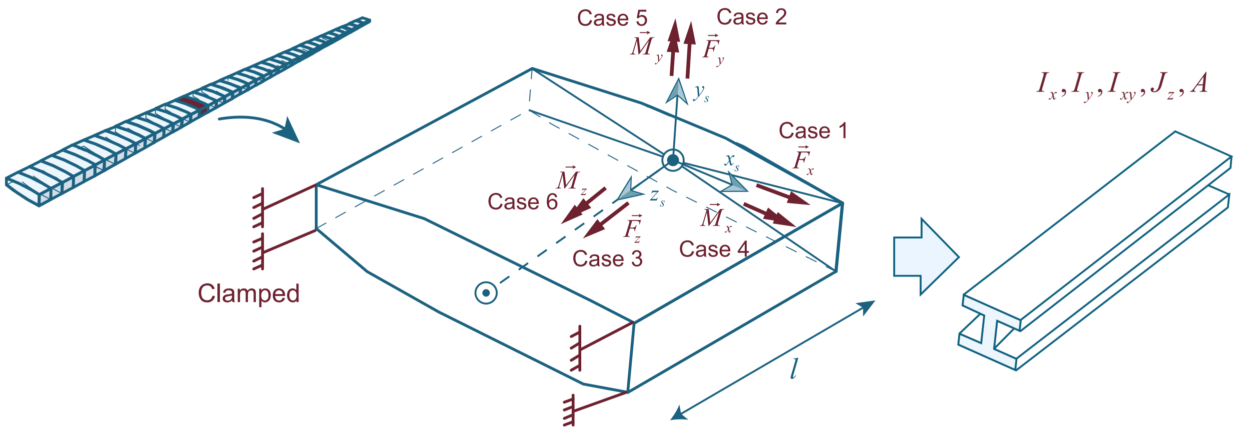


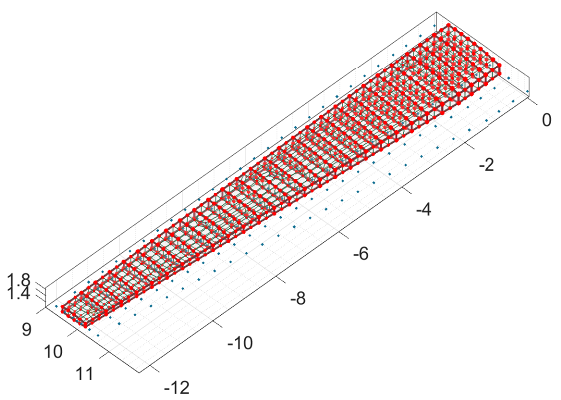
Figure 9: Geometry Condensation Process

2.6.3 Mass Condensation

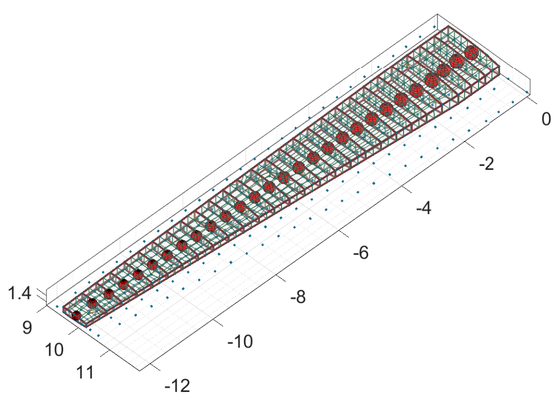
In the last step of the condensation process, the distributed optimum and non-optimum masses are lumped in point masses at each rib station. These point masses are connected to the beam grid points using rigid elements (RBE2).

The wing load-carrying masses are the masses that are calculated by the design criteria and are the direct result of the global finite element model (GFEM) strength sizing process. These masses are distributed over the shell and beam elements. To calculate the point masses of the finite element model, the mass matrix is extracted from a free-free structure analysis of the GFEM using Nastran DMIG (Direct Matrix Input) entry. The mass associated with each structural grid point in the GFEM is summed up according to closeness to the stick points, and the center of gravity and the corresponding mass for each rib segment is computed.

Figure 10a shows the initial mass distribution at shell grid points, and Figure 10b shows the masses lumped at beam grid points.



(a) Shell Model



(b) Beam Model

Figure 10: Mass Distribution of the Canti-Level Wing

The non-optimum structural masses (holes, brackets, cutouts, etc.), secondary structures masses (leading edge, trailing edge, flaps, etc.), and system masses (fuel system, pneumatic, etc.), powerplant masses (engine, nacelle, and pylon), and fuel masses are calculated according to the method that will be mentioned in the weight section, and the total weight distribution along the span is computed. The masses are lumped and applied to the structure according to the load cases.

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2.7 Sizing for Stability

For stability sizing of the strut for compressive load, the engineering method presented in [7] is implemented. In this method, the compressive load is extracted from the results of static analysis, and the required moment of inertia is computed. This moment of inertia is used to define the new cross-section of the strut. Based on the geometry of the strut, and the new cross-section, it is possible to calculate the wing penalty for the strut.

2.8 Structure Weight

The calculated weight from strength and stiffness requirements is the weight of primary load-carrying structures. To calculate the secondary items' weight, statistical and empirical methods can be used. In [7] a fixed fraction of take-off weight (4.43%), which was based on statistical analysis, is used for the secondary items. As this method does not consider the wing planform features, like flaps and slats, it was decided to use a more detailed method based on [22]. The weight breakdown and the associated method for computing the weight are presented in Figure 11.

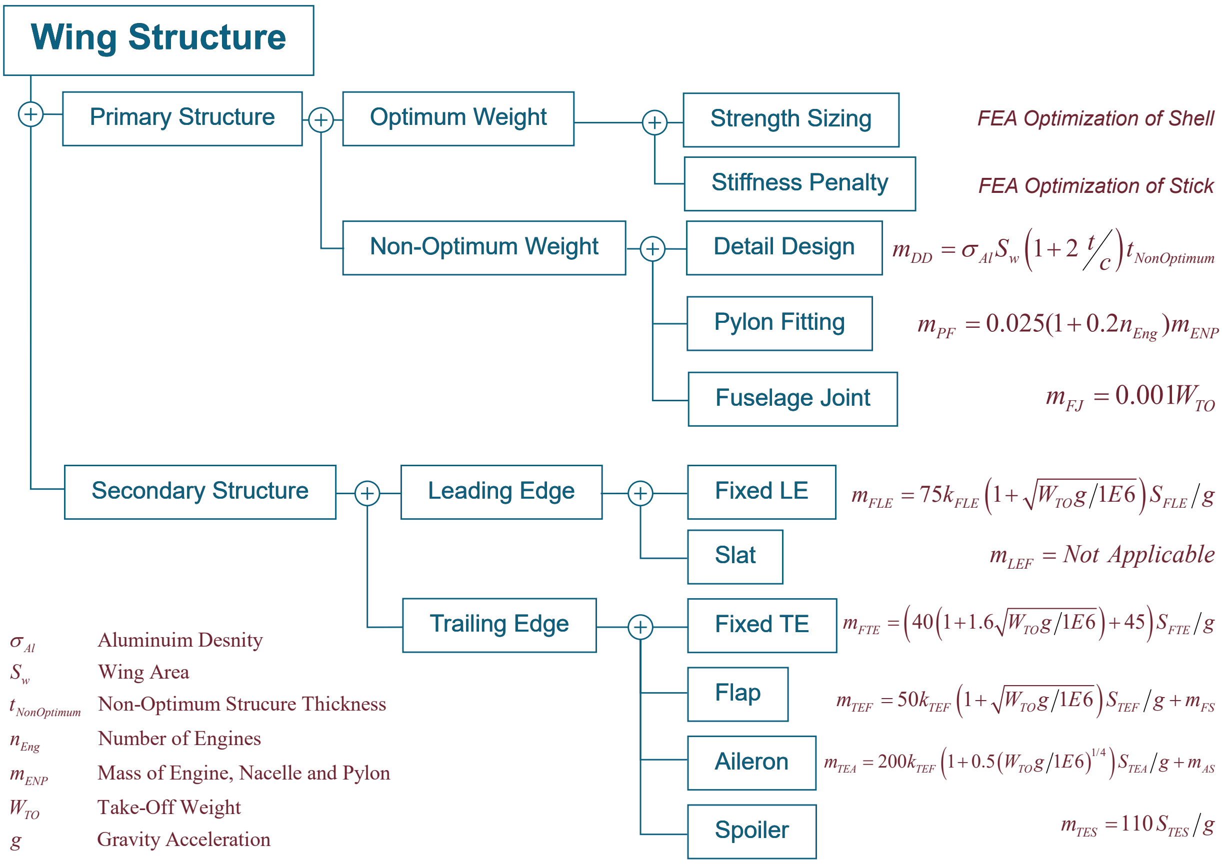

Figure 11: Geometry Condensation Process

Since the wing span of the truss-braced wing configuration is much larger than the conventional one, it may change the airport classification of the aircraft according to the International Civil Aviation Organization (ICAO) Annex 14 [23]. According to this standard, for the aircraft to be categorized under class C, the span should be between 24 [m] and up to but not including 36 [m]. For a greater span than the upper limit, a folding mechanism may be included, and this will induce a weight penalty. In this work, the weight penalty for the folding mechanism is calculated using the method provided in [24], which considered the shear force at the folding point, and it is validated using Boeing 777 folding wing weight penalty. The calculated weight is added to the wing structure, only if the span is greater than 36 [m]. The additional mass due to wing folding is presented in Figure 12.

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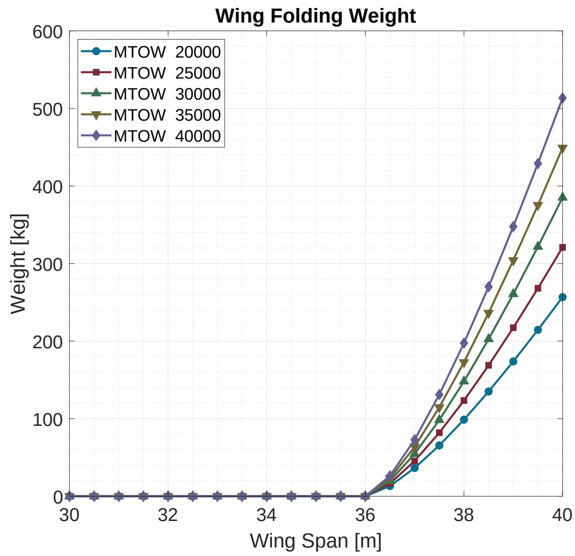
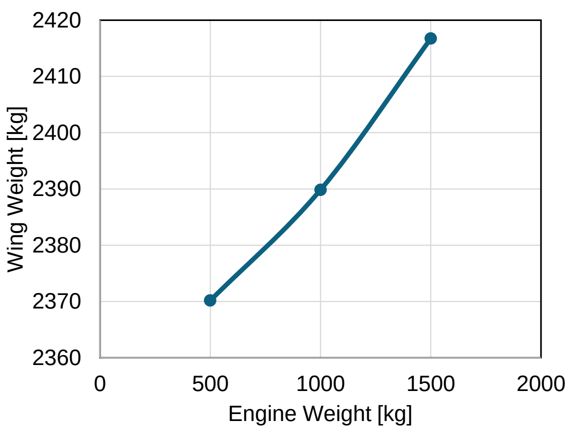


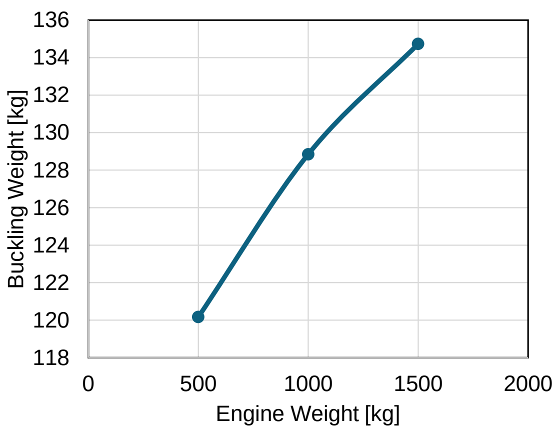
Figure 12: Wing Folding Weight

3. Results

The developed methodology is implemented to calculate the effect of the engine weight on the wing weight structure. The results is presented in Figure 13. As can be seen, as the engine weight is increased, the wing weight and the strut weight is increased.



(a) Total Wing Weight



(b) Strut Weights

Figure 13: Mass Distribution of the Canti-Level Wing

4. Conclusion

An automated framework for wing structure sizing and weight estimation is developed and implemented in MATLAB. This tool is capable of computing the weight of the primary wing structures using finite element analysis. The secondary items' weights are calculated using empirical methods. The developed methodology considers strength, stiffness, and stability requirements. The method is applied to the cantilever wing and truss-braced wing, and the results are presented. This method will be employed in a higher-level aircraft optimization tool, and the results show that by passing the wing planform information and flight conditions to this module, it is possible to calculate the wing weight.

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In future works, the stability calculations will be improved, and finite element methods will be used to size for stability.

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