Abstract

The objective of this paper is to obtain an optimal concept design of aircraft wing. Topology optimization is used to yield an optimized shape and material distribution for a set of loads and constraints within a given design space. Initially a solid wing was considered as design space and meshed using hexahedral elements to derive the optimal location of spars and ribs such that the structure has minimum compliance. Load data includes flight, fuel and engine loads. The wing was constrained in all degrees of freedom at the root. It was shown that the results can be used to derive an appropriate configuration similar to the existing design practices. A new geometry with spars, ribs, hinge locations and lag attachments was next considered. The skin was meshed with shell elements. Hinge loads were also included. The optimization gave a box section for front spar, channel section for rear spar, truss pattern for ribs and V shaped member for hinges. Using this global material distribution a component level topology optimization was performed. The concept design is realized into CAD and then a composite free-size optimization is performed to determine material distribution and ply drop regions etc. Finally a composite size and shape optimization is done.

Keywords: Aircraft Wing, Topology Optimization, Free Size Optimization, Composite Optimization

Introduction

The use of optimization techniques in engineering design has steadily been on the rise [1, 2]. Structural optimization has gained importance due to its significant impact in product development processes [3, 4]. A continuum approach to structural topology optimization was first introduced by Bendse and Kikuchi [5]. Optimization of Finite Element-based structures is acknowledged as a useful methodology for achieving important improvements in product design and is widely used in automotive and aerospace industries. Prior to commercial code development, conventional optimization in structures was achieved by determining the strain energy density and identifying material areas where a removal can be made or it is required to be strengthened and adopt a DOE approach. This is a tedious and time consuming process requiring skilled engineers. Many commercial Structural and Flow codes have now added optimization processes; however, topology optimization received considerable attention in recent times. One of the most popularly adopted codes for topology optimization is Altair OptiStruct [6].

Topology Optimization capability of OptiStruct has been successfully used in the recent times to obtain concept designs of aerospace such as Airbus 380 [7], A400M [8], Boeing 787 [9] etc. Topology optimization in OptiStruct uses SIMP (Solid Isotropic Material with Penalization) method which is also called density method [6]. In this method, the stiffness of the material is assumed to be linearly dependent on the density. The material density of each element is directly used as the design variable, and is normalized to have a value between 0 and 1 representing the state of void and solid, respectively.

The classical topology optimization setup involves the objective of minimizing the compliance with volume fraction as constraint. The compliance is the strain energy of the structure and can be considered a reciprocal measure for the stiffness of the structure. Volume fraction constraint specifies what fraction the volume is to be removed.
The remaining material is redistributed within the design space to obtain an optimal load path.

A typical aircraft wing consists of the following components: Spars, Ribs, Stringers, Skin and Flaps and Ailerons. Spars are the structural elements that ultimately bear the load carried by the wing. They run through the length of the wing from the point nearest to the fuselage out to the wing tip. Most wings have two spars; the front spar and the rear spar, but some wings may have as many as five. The leading edge and the trailing edge are additional structural components that run roughly parallel to the spars.

The ribs cross the spars and extend between the leading and trailing edges of the wing. Stringers are used in most cases for taking care of buckling. They run the length of the wing and may be above or below the spars and ribs. A skin covers the framework. Wings can be attached to the aircraft fuselage in different ways, depending on the strength of a wing’s internal structure. The cantilever wing structure is the strongest. It is attached directly to the fuselage without any external, stress-bearing structures. The semi cantilever wing has one or two supporting wires or struts extending between each wing and the fuselage. The externally braced wing, typically found on a biplane, has a number of struts and flying and landing wires.

Background of Topology Optimization

Topology optimization used in this paper is a combination of the Finite Element Method (FEM) with an optimization algorithm. As design parameters, every finite element gets a so called relative density \( \rho \), which may continuously vary between 0 and 1 and affects the elasticity tensor of a finite element as

\[
\frac{E}{E_0} = \left( \frac{\rho}{\rho_0} \right)^p
\]

where \( E_0 \) describes the nominal stiffness properties of the element. The task for the optimizer is to determine a density value for every element. The exponent \( p \) is a penalty parameter used to reach a result that is discrete as possible, by penalizing intermediate densities. If \( \rho \) tends to zero, the stiffness tends to zero too. This means, the element could be deleted because it is not important for the structure. If the density reaches a value of 1, the element is very important for the structure and may not be removed. This approach is called SIMP (Solid Isotropic Material with Penalization [10]).

To illustrate the capabilities of the topology optimization, a simple example problem is illustrated here: Fig.1 shows the meshed design-space of a "truss".

Here the entire space is considered design space. The task is to find a structure with a minimum volume and that the maximum displacement remains same as baseline. The baseline result is shown in Fig.2 with \( E = E_0 = 210 \) GPa, the maximum deflection at the node shown is 0.001826 mm. After three iterations the density distribution or elasticity distribution as displayed in Fig.3 is arrived at. The "red elements" indicate a density = 1. This means these elements are very important and have their nominal stiffness properties. The "blue elements" have a density close to zero and therefore a very low stiffness and they can be removed. We can build the structure around the "red elements".

Altair OptiStruct is a finite element and multi-body dynamics software for the design, analysis, and optimization of linear structures. The optimization problem can be stated as

\[
\text{Min } f(X) = f(X_1, X_2, \ldots, X_n) \quad \text{subject to}
\]

\[
g_j(X) \leq 0, \quad j = 1, \ldots, m
\]

\[
X_i^L \leq X_i \leq X_i^U, \quad j = 1, \ldots, n
\]

where \( f(X) \) is the objective function, \( g(X) \) are the constraints, both of which are functions of the design variables. There are \( m \) constraints and \( n \) design variables. The type of design variables can be size, shape/topography, and topology.

Topology optimization is a mathematical technique that generates an optimized shape and material distribution for a set of loads and constraints within a given design space. OptiStruct algorithm alters the material distribution to optimize the user-defined objective under given constraints. It can be introduced at the initial design step to enhance the design process. Chart 1 shows an optimization driven design process where topology optimization is used as an idea generator in the concept phase [11].

The design space can be defined using shell or solid elements, or both. The classical topology optimization can be set up solving the minimum compliance problem as well as the dual formulation with multiple constraints. Manufacturing constraints can be imposed using a mini-
mum member size constraint, draw direction constraints, extrusion constraints, symmetry planes, pattern grouping, and pattern repetition.

OptiStruct first performs a baseline linear analysis. For basic linear analysis types, gradient based optimization is the most efficient approach. This method may not work well for truly discrete design variables, such as those that would be encountered when optimizing composite stacking sequences. However, the method has been adopted for discrete design variables where the discrete values have a continuous trend, such as when a sheet material is provided with a range of thicknesses. The adopted method works best when the discrete intervals are small. In other words, the more continuous-like the design problem behaves, the more reliable the discrete solution will be. For example, satisfactory performance should not be expected if a thickness variable is given two discrete values 0 and T [6]. The number of constraints whose gradients need to be calculated for, can be reduced using a process called constraint screening. It is the process by which the number of responses in the optimization problem is trimmed to a representative set. With constraint screening, constraints that are not close to being violated are ignored. This reduces the number of active constraints, and therefore the sensitivity costs.

After constraint screening is completed, the code automatically performs a sensitivity analysis to calculate the gradient. An approximation method is used in constructing an analysis system which in turn reduces the number of finite element analyses to be performed. The optimization algorithm is used in conjunction with approximate method to determine approximate optimum design. This design is then analyzed using a full finite element analysis, which is used as the basis for the next approximate optimization problem. This process is continued until convergence is achieved as illustrated in Chart 2. This method is also used for shape and weight optimization of rotating structures [12]. For composite optimization, reference may be made to Grdal, Haftka and Hajela [13].

Post World War II Wing Designs

Civil aircraft manufactured by Boeing, Airbus and McDonnell Douglas show that the layouts have been extremely stable in the post-World War-II era, with no dramatic variation between manufacturers, and between models of the same manufacturer [14]. Wings made by these companies are based on the same few-spars-many-ribs design, with near-constant rib spacing, as may be seen in Figs.4a and 4b. While most of the designs contain only two spars, a few include an additional mid-spar that goes partway, from fuselage up to wing-mounted engine.

An interesting variation is observed among some German aircraft dating back to the pre-World War days. The old Heinkel bomber and the current Dornier aircraft both utilize iso-truss construction in the wing [15] as shown in Fig.5.

Here, we illustrate the topology optimization of an aircraft wing beginning with a fully solid wing to understand how this method allocates the material distribution with suggestions for removal and how such a structure resembles the design practices in place since World War II.

Topology Optimization of Solid Wing

A Solid wing made of Aluminum with Wing Span 13.5 m, Constant chord wing span 4.74 m, Tapered chord wing span 8.76 m shown in Fig.6 is taken. The Chord length at wing root is 2.68 m and at wing tip 1.39 m. Material properties are taken as Youngs Modulus = 70000 MPa, Poissons ratio = 0.33 and Density = 2.60 x 10^{-09} tonne/mm^3. The loads taken on the wing include flight load, fuel load and engine load. The airfoil sections, where pressure values are determined by Vortex Lattice method using XFOIL (a public domain code) are given in Fig.7. The pressure values at stations every 500mm apart the wing span are given in Fig.8.

Fuel tank starts from 20% and ends at 85% of constant chord wing span, see Fig.9. Center of Gravity of the fuel tank is at 1.27m from leading edge and 2.5m from wing root. Fuel load is 1730 Kg per wing and is distributed through RBE3 element.

The engine is mounted from 85% to 98% of constant chord wing span; see Fig.10. Center of Gravity of the fuel tank is at 1.27m from leading edge and 2.5m from wing root. Fuel load is 1730 Kg per wing and is distributed through RBE3 element.

The engine is mounted from 85% to 98% of constant chord wing span; see Fig.10 and its weight is 800 Kg. It is assumed to be mounted on front and rear spars through two ribs and the weight distribution at the four nodes of intersection between the two spars and ribs shown in Fig.10 is given in Table-1.

The wing was constrained in degrees of freedom analogous to a cantilever beam. All the nodes in the root tip face of the airfoil section were constrained, see Fig.11.
Analysis and Optimization

The starting point for any OptiStruct optimization is a working OptiStruct finite element model. Therefore a linear static analysis was performed and the results were checked prior to setting up an optimization problem. (It may be noted here that the baseline is a solid wing which will be heavy and rigid and the results obtained here are of no consequence and hence not presented here. The analysis and optimization will however go through this step in the first instance.). The next step is to setup an optimization problem on the finite element model using OptiStruct. Material density of each element is a design variable which is defined using DTPL (Design Variables for Topology) card. In the present case, two responses from analysis (compliance and volume fraction) are defined using DRESP1 in which volume fraction (fraction of volume to be removed) is defined as constraint and compliance (inverse of stiffness) is defined as the objective function. Three design cases were formulated and discussed below.

Design Case 1: In this design case, a classical topology optimization was performed without any manufacturing constraints. The classical setup involves subjecting the model to the objective of minimizing the compliance with volume fraction as constraint. Fig.12 shows material density plot obtained from the run and Fig.13 shows iso-density boundary plot for the elements above density 0.2. Fig.14 shows how the objective function changed with respect to the iterations (41 in this case) until convergence is achieved.

It can be observed from Fig.13 that the resulting structure already resembles somewhat a classical wing type construction that can be adopted in the concept. We also observe from bird wings, how nature has designed them as feathy structures to reduce weight and provide minimum effort for propulsion. Optimization process leads to similar designs that nature provides. Here we have not constrained airfoil skin shape and a non-design space criterion is adopted in design case 2.

Design Case 2: In this design case, one layer of boundary elements was taken out of the design space to represent skin, see Fig.15. Extrusion manufacturing constraint was added to derive the location of spars. Extrusion constraint is used when it is desirable to produce a design characterized by a constant cross-section along a given path. Fig.16 shows iso-density boundary plot for the elements above density 0.3.

We can observe from Fig.16 that a single spar formation is suggested by the optimization process. To see whether a conventional 2 spar configuration is realized, another design case is considered by splitting the wing into three regions.

Design Case 3: For this design case, the design region was split in to three and extrusion manufacturing constraints were defined for each design region. The result was a two spar configuration, shown in Fig.17.

From these three case studies, we see that a conventional two spar configuration of a wing with several ribs as conceptualized from intuition and design practices followed right from the beginning of aircraft wing design since World War II is derivable through optimization. This gives us confidence of utilizing this powerful optimization technique for weight reduction of the existing designs or a minimum weight design that can be achieved for new designs.

Topology Optimization of Wing with Proposed Geometry

In this study, a new geometry with spars, ribs, stringers, hinge locations and lug attachments was considered. The objective was to derive material layout for ribs, hinge locations, stringers and cross section for front and rear spars. The given geometry, shown in Fig.18 was only up to rear spar location and did not include flaps and ailerons. It has two spars viz. the front spar and the rear spar. There are two fuel tanks, one at the end of constant chord wing span and the other in tapered chord wing span. There are three closely spaced ribs at the end of constant chord span for mounting the engine. It has six hinge locations attached to the rear spar and four lug attachments for mounting the wing to the fuselage bulkhead.

The given load on the wing includes flight load, fuel load, engine load and hinge loads (flap and aileron loads). Aircraft cruise condition was obtained for the flight load using Vortex Lattice method. Pressure distribution on the
pressure surface is shown in Fig.19. The Planform of the wing was divided in to 200 panels and the pressure distribution was constant within each panel giving a discrete distribution. Planform from rear spar to the leading edge was divided in to seven panels. The discrete distribution of pressure on each panel from rear spar to leading edge is shown in Fig.20 and the division of panels in Planform is shown in Fig.21.

There are two fuel tanks. The capacity of the fuel tank is 2500 liters. The fuel tank at constant chord span takes 1000 liters and the fuel tank at tapered wing takes 1500 liters. The load was calculated using kerosene properties and distributed using RBE3 element see Fig.22.

Engine (800 Kg) is assumed to be mounted on both front and rear spar along with three closely placed ribs at the end of constant chord wing span. Front spar takes 500 Kg and the rear spar takes 300 Kg. The load is distributed as shown in the Table-2. The weight distribution at the six nodes of intersection between the two spars and ribs is shown in Fig.23 as given in Table-2.

<table>
<thead>
<tr>
<th>Table-2 : Engine Load</th>
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<tbody>
<tr>
<td>First Rib</td>
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<tr>
<td>Front Spar</td>
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<tr>
<td>Rear Spar</td>
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Figure 27 suggests predominantly V-shaped members for hinges, as usually practiced. Material density and iso-density plots for hinges are shown in Fig.28 and Fig.29. Figs.28 and 29 suggest a truss pattern for the ribs, once again a general practice adopted in the design of ribs. The regions in red are non-design space. Finally, the iso-density plot for stringers is shown in Fig.30. This Iso-density plot suggests L-shaped cross section for stringers around the rear spar. Two stringers at the leading edge which are of not much significance in providing stiffness to the model are removed.

**Composite Optimization**

After obtaining concept design from isotropic topology optimization, the next phase is to do a composite optimization. The concept design from topology optimization is realized in to CAD. Then a composite free-size optimization is performed to determine material distribution and ply drop regions etc. Finally a composite size and shape optimization is done.

Composite optimization is done in three phases. They are:
- Free-sizing optimization
- Sizing optimization
- Ply-stacking optimization

Composite free-sizing optimization is to create design concepts that utilize all the potentials of a composite structure where both structure and material can be designed simultaneously. By varying the thickness of each ply with a particular fiber orientation for every element, the total laminate thickness can change ‘continuously’ throughout the structure, and at the same time, the optimal composition of the composite laminate at every point (element) is achieved simultaneously. Manufacturing constraints like lower and upper bound thickness on the laminate, individual orientations and thickness balance between two given orientations can be defined.

Sizing optimization is performed to control the thickness of each ply bundle, while considering all design responses and optional manufacturing constraints. Ply thicknesses are directly selected as design variables.

Composite plies are shuffled to determine the optimal stacking sequence for the given design optimization problem while also satisfying manufacturing constraints like control on number of successive plies of same orientation, pairing 45° and -45° orientations together etc. An example
problem is illustrated below to convey the concepts of methodology. This method has been successfully applied to production parts on current aircraft designs. In this problem, optimization was performed on a composite I-Beam with the objective of minimizing weight. The constraints taken are

- Buckling parameter $\lambda > 1$ (no buckling)
- Maximum strain $\varepsilon < \varepsilon_{\text{allow}}$ (max. strain at failure)
- Maximum deflection $\delta < 12.7$ mm

The beam is 3048 mm long and fixed at both the ends. It is subjected to a distributed load of 0.7 MPa as shown in Fig. 31. It is constructed by Top/Bottom Cap and Web C-Channel Laminate Method, shown in Fig. 32. The beam can have ply drops along the length but Top and Bottom Laminate must be symmetric at all Cross-Sections.

Composite free-size optimization was performed to obtain the material distribution and define ply drop regions based on the results. The input model for base laminate for web and cap plies (8 layers each 2.54 mm thick) is chosen with 4 ply orientations of convenience and carbon-epoxy material (hybrids can also be used) as given in Table-3. The minimum and maximum thicknesses can also be provided as constraints. In this work no such constraints were stipulated. The results for the four plies of free-sizing optimization are given in Fig. 33a to 33d.

Then a size optimization was performed to determine the thickness and laminate family within the region. Based on this result, the beam was divided into regions for size optimization as shown in Fig. 34. The model built on free-size interpretation is shown in Fig. 35. All the plies are assumed to have 2.54 mm thickness in all regions as given in Table-4a for TopCap_1.

The size optimization result is given in Fig. 36 and Table-4b. The total ply bundle thickness is only 1.297 mm for TopCap_1 shown in the extreme left end of the beam. Then a size optimization was performed on the model, results of which are shown in Fig. 36. Laminate family determination was done based on the size optimization results. This can be output in excel spreadsheet containing details of laminate family for each region. The final model derived is checked to ensure whether it meets all the design objectives. The results of the analysis performed on the final design are shown in Fig. 37.

Overview of composite optimization described here is summarized in Fig. 38. Design of composite structures is now increasingly becoming a common approach, e.g., [16]. The methodology discussed above has been successfully adopted in composite optimization of the 787 Dreamliner Horizontal Stabilizer CFRP Composite Main Box [9] and Weight Reduction of a Formula One Composite Component [17].

Conclusions

The importance of topology optimization in obtaining concept design and the capability of composite optimization is demonstrated. This concept design has to be realized in to CAD. Size/composite optimization can then be performed to obtain a final design.
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References


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Fig. 1 Topology Problem for a Truss

Fig. 2 Baseline Result

Fig. 3 Distribution of Elasticity after Topology Optimization

Chart 1: Optimization Driven Design Process

Chart 2: Optimization Process

Fig. 4a Spacing of Spars in Civil Transport Wings

Fig. 4b Spacing of Ribs in Civil Transport Wings
Fig.14 Design Case 1 - Objective Function Vs Iterations

Fig.15 Design Case 2

Fig.16 Design Case 2 - Iso-Density Plot above 0.3 Density

Fig.17 Design Case 3 - Iso-Density Plot above 0.3 Density

Fig.18 Proposed Wing Design

Fig.19 Pressure Distribution

Fig.20 Discrete Pressure Distribution

Fig.21 Panels Division in Planform

Fig.22 Fuel Tanks
Fig. 23 Engine Load

Fig. 24 Boundary Conditions

Fig. 25 Material Density Plot - Spars

Fig. 26 Iso-Density Plot - Spars (Density above 0.3)

Fig. 27 Iso-Density Plot - Hinges (Density above 0.3)

Fig. 28 Material Density Plot - Mid Rib

Fig. 29 Iso-Density Plot - Ribs (Density above 0.3)
Fig. 30 Iso-Density Plot - Ribs (Density above 0.3)

Fig. 31 I-Beam Model

Fig. 32 Cross-Section of I-Beam Model

Fig. 33 Free Size Results of Plies
Fig. 34 Free-Size to Gauge Optimization

Fig. 36 Size Optimization Result

Fig. 37 Analysis Results of Final Design

Fig. 38 Overview of Composite Optimization Methodology