WEIGHT OPTIMIZATION OF AN AIRCRAFT WING COMPOSITE RIB USING FINITE ELEMENT METHOD

UÇAK KANADINDAKİ KOMPOZİT RİB PARÇANIN SONLU ELEMANLAR YÖNTEMİ KULLANILARAK AĞIRLIK OPTİMİZASYONU UYGULAMASI

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To my precious family..

ÖZET

UÇAK KANADINDAKİ KOMPOZİT RİB PARÇANIN SONLU ELEMANLAR YÖNTEMİ KULLANILARAK AĞIRLIK OPTİMİZASYONU UYGULAMASI

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Bu çalışmada uçak kanatlarında kullanılan ve temel yapısal parçalardan birisi olan kompozit rib parçaların ağırlık optimizasyonu ve ağırlık azaltma prosedürü üzerine çalışılmıştır. Kompozit rib parçanın üzerinde yapılan çalışmada katman sayısı, kalınlık ve kompozit katman dizilimi optimize edilerek, optimal ağırlıktaki tasarım elde edilmiştir. Elde edilmiş bu optimize tasarım ile birlikte uçak kanadının temel yapısal analizleri tekrarlanmış ve optimal tasarımın verimliliği gözlemlenmiştir.

Bu çalışma kapsamında çoğu uçak kanadı tasarımında kullanılan temel tasarım yöntemleri ve varsayımları kullanılarak bir uçak kanadı modeli oluşturulmuştur. Literatürdeki benzer çalışmalarda kullanılan modellerinin kaba geometrik ölçülerinden faydalanılarak kanat modeli boyutlandırılmış ve bu dış geometrinin üzerine temel yapısal parçalar konumlandırılmıştır. Analizler ve optimizasyon çalışmaları tasarlanan bu prototip uçak kanadı üzerinde gerçekleştirilmiştir. Tam uçak kanat modeli elde edildikten sonra sonlu elemanlar modeli oluşturularak belirlenen bir uçuş koşulu

sırasında kanat yapısal elemanlarının maruz kalacağı yük dağılımı hesaplanmıştır. Elde edilen bu yük dağılımı kullanılarak tasarım aşamasında rib parçalarına verilen başlangıç kalınlıkları optimize edilmiştir. Optimizasyon aşamasında sonlu elemanlar modelinden elde edilen yükler kullanılarak önceden belirlenmiş her bir bölge için güvenlik katsayısı hesaplanmış ve bölgesel olarak gereğinden fazla kalın tasarlanmış bölgelerde ağırlık azaltma çalışması yapılmıştır. Bu metod HyperSizer programı kullanılarak uygulanmıştır.

Bahsi geçen optimizasyon yöntemini kullanılarak optimal katman sayısı ve katman dizilimi elde edilmiş, gereğinden kalın tasarlanmış yapılar optimize edilmiş ve böylece ağırlık azaltımı sağlanmıştır. Sonuç olarak, düzenli kalınlık dağılımına sahip rib tasarımına göre daha hafif ve benzer mukavemet değerlerine sahip bir rib modeli elde edilmiştir. Bu çalışma kapsamında önerilen yöntem, tasarım sürecinin başında uçağın temel ağırlık analizi çalışmaları esnasında oldukça kullanışlıdır.

Anahtar Kelimeler: Ağırlık Optimizasyonu, Rib, Kompozit Ağırlık Optimizasyonu, Sonlu Elemanlar Yöntemi

ABSTRACT

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In present work, weight optimization and weight reduction of composite wing rib design which is one of the wing structural components used in aircraft wing design are studied. As a result of the optimization study on composite wing rib, number of ply and stacking sequence are optimized and optimal weight of the design is obtained. In addition, structural analysis methods are applied on the optimal aircraft structural wing rib design, obtained as a result of the optimization study, in order to examine efficiency of the design in terms of total deflection effects on the wing design and weight reduction.

During present study, a hypothetical aircraft wing design is modeled in accordance with the well-known aircraft wing design standards and assumptions. Outer geometry of the wing design is obtained exploiting common wing models of the similar studies in the literature. As the outer geometry of the wing is designed, common structural wing components are placed into model accordingly. Structural analysis and optimization methods are studied on this aircraft wing design. After obtaining overall wing design, a

finite element model of the design is generated with the presumed aircraft load case and load distribution on the aircraft wing is obtained. Initial thickness of the rib design is optimized by using load distribution obtained as a result of finite element analysis. During optimization study, margin of safety for all predefined regions are investigated and the regions which are evaluated as over safe are determined. The thickness of the regions that are unnecessarily thick and over safe are optimized locally. This optimization method is carried out by using HyperSizer software.

By using this optimization method, optimal ply number and stacking sequence are obtained. Over safe designed regions are optimized in terms of weight and thickness. A lighter aircraft wing rib design compared to the initial uniform thickness design is obtained with nearly same stiffness values is reached.

This method is important and beneficial especially at the first stage of the design process since it gives an idea about foreseen design can be able to satisfy design criteria such as weight or not. Optimization method described in this study is beneficial to examine total weight of draft aircraft design alternatives at early stages of the design process.

Keywords: Structural Optimization, Rib, Composite Weight Optimization, Finite Element Method

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I must express my very deep gratitude to my family for their endless support. I express my thanks to my mother, İksir Kandemir. Without you, I could have not succeeded anything. And my brother, Görkem Kandemir, thanks for being with me all the time and supporting me in every single possible way through my life. Your precious advices helped me to find my way in my professional career. My special thanks go to my wife Bengü Kandemir for her everlasting support and love through my life. Her support and advice turned this study into a pleasure.

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LIST OF SYMBOLS

P	:	pressure
ρ	:	density
V	:	velocity
y	:	height
t	:	thickness
c	:	chord length
σ	:	normal stress
τ	:	shear stress
m	:	mass
g	:	gravitational acceleration
gsm	:	gram per square meter

LIST OF ABBREVIATIONS

CAE	Computer Aided Engineering
CFRP	Carbon Fibre Reinforced Polymer
DOF	Degree of Freedom
FE	Finite Element
FEA	Finite Element Analysis
FEM	Finite Element Model
LE	Leading Edge
MoS	Margin of Safety
NASA	National Aeronautics and Space Administration
OB	Outboard
SF	Safety Factor
SPC	Single Point Constraint
TE	Trailing Edge
UD	Unidirectional
IB	Inboard

1 INTRODUCTION

1.1 Problem Definition

In modern aviation industry, weight of the structural design plays an important role due to the fact that over weighted designs are more expensive to operate and lack of flight ranges compared to the lighter designs. In order to have an efficient and beneficial structural design, weight should be kept as minimum as possible maintaining structural strength. However, minimizing weight of a structural design causes the strength of the design decreases. Therefore, structural weight optimization is a key concept to obtain weight efficient structural design for airborne platforms.

As shown in Figure 1, aircraft designs comprised of components which have certain duties to perform. Wings are the aircraft components which are expected to generate lift force in order to keep aircraft flying. Two separate wings are designed to produce lift force. As a result of the lift force acting on the wings, huge amount of deflection and stress are experienced by the structural elements of wing. This phenomenon makes wing structural components safety critical. These critical structural components are designed over safe in most of the case, resulting with a wing which is much heavier that it is expected to be. In the industrial design stage, most of the case, this criticality brings oversafe and heavy component designs. From this perspective, weight and thickness optimization of structural wing components become more important in order to have lighter airborne platform designs. Furthermore, in the case of having composite structural elements, stacking sequence is important topics to be studied during the weight optimization.

Most of the aerospace industrial designs are composed of the structural elements which are designed by considering the maximum stress or the maximum strain on the element, and this design approach concludes oversized and over weighted structural elements. Instead of considering only the maximum load values on a structural element, analyzing the load distribution contributes weight optimization of both structural elements and the overall airborne platform. Each individual predefined local area on the structural component has its own maximum stress and maximum strain values. Therefore, each individual local region have their own thickness and stacking requirements which implies that it is possible to have more weight-efficient structural designs. This study focuses to describe a design methodology which is used to have optimal structural elements in weight, maintaining the stiffness characteristics of the structural elements.

Common type of aircraft wings are composed of certain types of structural components such as upper and lower skin panels, spars and ribs which may be seen in Figure 2. Upper and lower panel are structural components of an aircraft wing that maintain the aerodynamic consistency. Generated necessary lift force to be able to have smooth flight condition of aircraft is acquired with the help of those upper and lower skin panels. When the air flows from upper and lower side of the wing, pressure difference is occurred with the help of airfoil shape of the wing. This pressure difference obeys the Bernoulli principle as shown in equation (1).

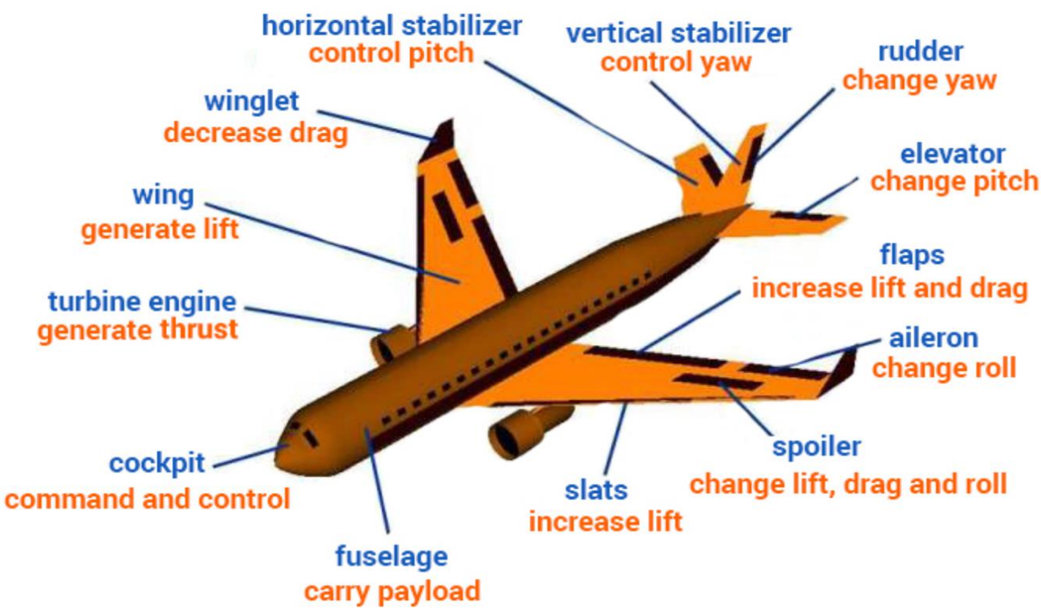


Figure 1. Aircraft common components [1]

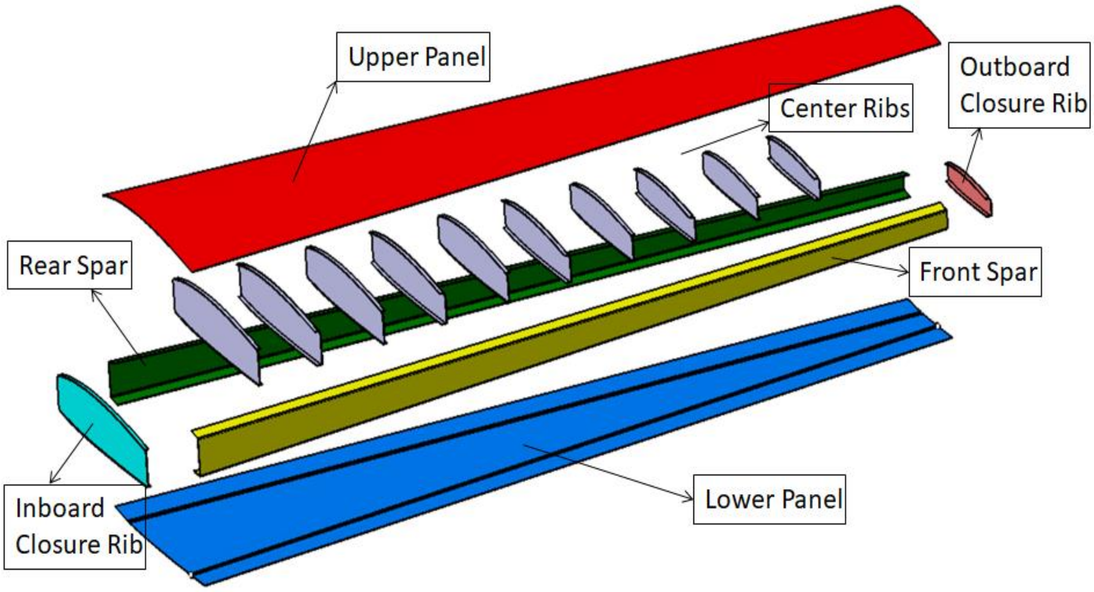


Figure 2. Main structural components of an aircraft wing

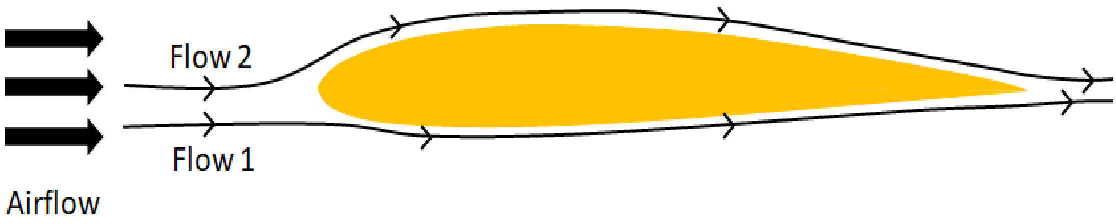


Figure 3. Airfoil flow diagram

$$p_1 + \frac{1}{2}\rho {V_1}^2 + g\rho y_1 = p_2 + \frac{1}{2}\rho {V_2}^2 + g\rho y_2 \quad (1)$$

Where; p : pressure,

ρ : density of the air,

V: airflow velocity,

y : height

g : gravitational acceleration

Since the velocity of flow 2 is greater than flow 1 due to geometry of the airfoil, pressure is higher at the lower side, therefore wing is pushed upwards. As a result, upper and lower panel are designed to produce enough lift force to overcome total gross weight of the aircraft.

Spars are another main structural parts placed in a wing in order to have the wing stiffer and to defend the aircraft wing against bending loads. Number and types of spars (Z spar, C spar etc.) are design issues which are needed to be decided during design of the aircraft wing

Ribs are one of the main structural components of the aircraft wings as well. Similar to spars, ribs are used in the wing design to make it stiffer. However, ribs are used to defend the wing against shear loads whereas spars are used against bending forces. Rib is the structural component in the wing design highest in number. Number of rib component in the wing design is at least two and increases in accordance with the load distribution.

As mentioned above, rib is one of the most important structural parts that common aircraft wing models have. Since ribs are high in number in the wing design, optimizing rib designs would make possible weight reduction in the overall design easily. Also load distribution of the rib components are more complicated than the other structural parts. According to common load case, stress values goes down from wing root to wing tip. Therefore, it is clear that wing root part of the skin panels and spars should be designed thicker. On the other hand, this situation is not the same in rib components since load distribution can vary much more compared to the other components. As a result of the reasons mentioned above, rib is the most suitable structural element whose weight may be studied to reduce weight of the overall aircraft structure.

As mentioned previously, structural design engineers in the aviation industry size the components according to maximum overall stress or strain on the part. However, in some cases, this method induces over safe and over weighted designs. On the contrary, the method studied in this thesis proposes to size the structural composite components according to load distribution on each component individually. By applying this method, it would be possible to have lighter and efficient designs with the approximately same stiffness values. In the proposed method, as shown in Figure 4, thicknesses on the regions that have low level of stress are reduced in order to keep component weight lower. Stacking sequence is also optimized and so that more efficient, lighter and beneficial aircraft wing component is obtained.

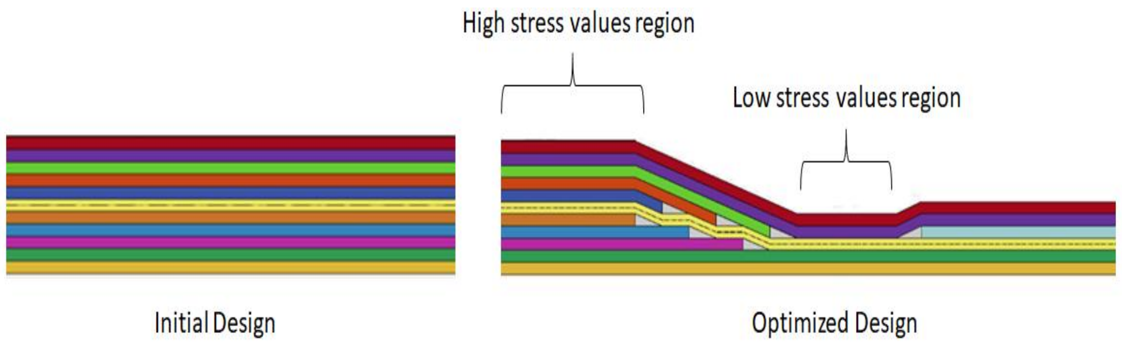


Figure 4. Optimization schematic

1.2 Aim and Scope of the Thesis

Main aim of this thesis is to develop a method for obtaining weight optimized aircraft structural designs and getting weight efficient designs. To be able to have efficient aircraft composite structural design, ply number (i.e. thickness) and ply stacking sequence optimization are optimized. This optimization method is applied on an aircraft composite structural rib part. Firstly, a new exemplarily aircraft wing design is generated. Then, again a selected exemplarily load case is used and finite element model is generated. By using this finite element model of the aircraft wing, load distribution and load paths are obtained. Optimization method is applied according to rib loads that are obtained from this finite element model.

In the optimization stage, Hypersizer software tool is used. HyperSizer is computer-aided engineering (CAE) software used for analysis and optimization of both metallic and composite structural parts. It is originally developed at the US National Aeronautics and Space Administration (NASA). It is used in this study as a sizing tool. Finite element model that is created in this work and results file of it are fed as inputs to the Hypersizer. It iterates the model and obtains optimal ply number and optimal stacking sequence by considering producibility of the suggested design solutions.

Main objective of this work can be summarized as make possible to generate weight efficient aircraft composite structural rib by using optimization method as described. With this method, engineers have a possibility to know that which parts of the structural parts experience with the bigger amount of load and which sections of the structural parts should be design thicker or thinner. Moreover, most efficient stacking sequence could be known in the design stage. By doing so, lighter and almost equally stiff structural aircraft composite designs are available in the industrial applications. Lighter and stiff structural design is very important in the especially aviation industry and engineers could be able to obtain those efficient structural products in the design stage.

1.3 Outline of the Thesis

The outline of the thesis is as follows: In the second chapter, conducted literature survey is presented. Researches about similar studies are mentioned. Results of the similar works are investigated and which aspects of them could make advantages in this thesis are observed. This chapter includes investigation topics of aircraft wing design and structural optimization as well.

The third chapter provides the theoretical background which is relevant to work in this thesis. Main aircraft design theories are explained and fundamentals of finite element method are mentioned. Thickness optimization and stacking optimization methodologies are explained. This section includes how Hypersizer software works in optimization stage.

Main aircraft wing box design methodology used in present work is mentioned in fourth chapter. Airfoil details used in this study is explained and geometry and material of wing box design is explained in this chapter.

Fifth chapter explains the finite element analysis of the initial wing box design. All details of the finite element model created for the analysis is explained in this chapter. Analysis results are presented at the end of the chapter.

Sixth chapter describes the weight optimization methodology for the selected composite wing rib. Optimized thickness and composite stacking sequence of the designed aircraft composite wing rib is presented in this section.

Finite element analysis of the optimized wing box design is presented in the seventh chapter. Effects of applied optimization process on the overall wing box model are discussed in this chapter.

The eighth chapter is comprised of conclusions and discussions. In this chapter, main outcome of the present study is explained. Results of the study are discussed and compared with the initial design in terms of performance and weight. Also, possible future work is mentioned in this section.

2 LITERATURE SURVEY

2.1 Introduction

In this part of the thesis, pioneering and leading studies related to composite components' weight optimization procedure are mentioned. Previous literature is introduced as two main concepts, which are related to aircraft wing box design and weight optimization. At the end of this chapter, brief summary of the chapter is provided.

In the first section, some of the researches related to aircraft wing design are investigated including their implementations to this study. In the second section, weight optimization concept is focused mainly. Optimization methods applied on the similar cases are retrieved and discussed in this section. Applicability of those similar works from different fields in the industry is examined. Best optimization ways to obtain weight efficient aircraft composite designs are investigated and summarized in this section.

2.2 Aircraft Wing Box Design Concept

This study aims to generate weight efficient composite aircraft part design and optimize prespecified aircraft wing composite wing rib. To be able to perform such optimization procedure, load distribution on the part should be obtained in exemplarily selected load case. To do so, finite element model of the wing should be generated.

Having well-known and commercialized aircraft wing model may not be acquired with various reasons. This point brings us to develop our own aircraft wing box design. So, literature related to aircraft wing design is examined and summarized carefully in this chapter.

As mentioned before, aircraft wings have certain types of structural components such as upper and lower panels, spars, ribs and riblets. Since it affects many parameters, first action in the aircraft wing design is to choose airfoil type. Airfoil type of the aircraft wing is selected in accordance with the aircraft design requirements such as gross weight of the aircraft, maximum cruise speed, takeoff and landing distance, stall speed and overall dynamic efficiency etc. [2].

Figure 5 illustrates the key parameters of a standard airfoil type. All those key geometrical parameters should be identified in the first stage of aircraft wing design according to project requirements. On the other hand, apparently such procedures need huge research and development processes including design and tests in huge wind tunnels. In this case, a variety of airfoil families play an important role for the aircraft wing designers as shown in Figure 6.

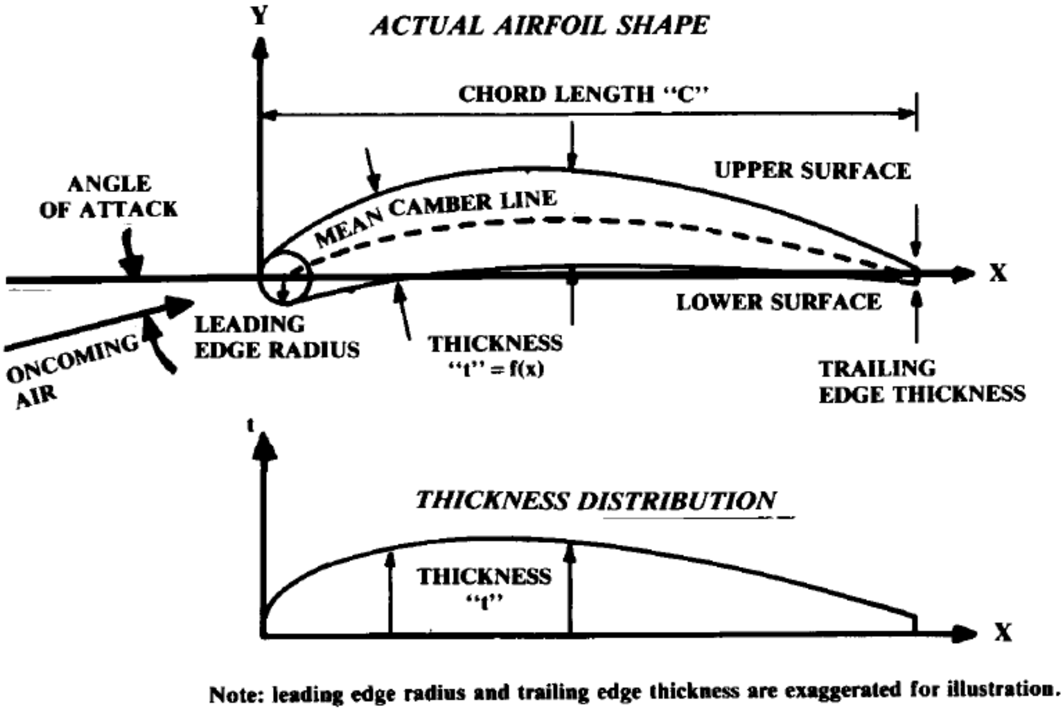


Figure 5. Airfoil geometry parameters [2]

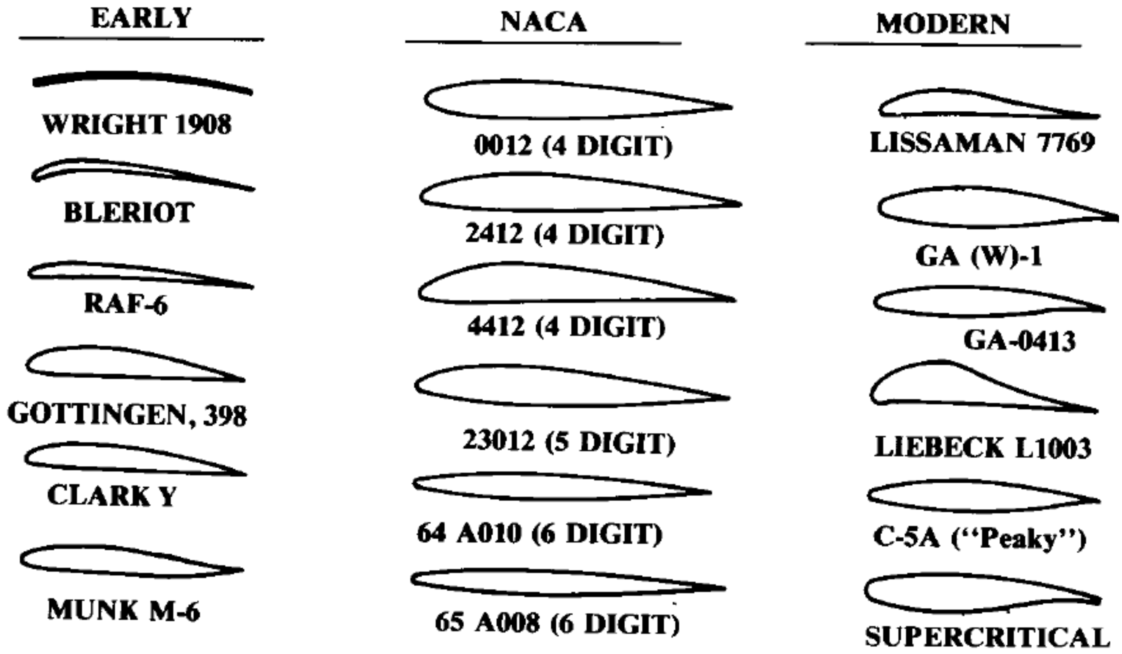


Figure 6. Typical airfoil families [2]

Choosing airfoil type from catalogs considering factors such as airfoil drag during cruise, stall and pitching-moment characteristics, the thickness available for structure and fuel and the ease of manufacture is an old-fashion method. Today's modern applications are based upon inverse computational solutions for desired pressure or velocity distribution on the airfoil. However, choosing airfoil type from catalogs is still applicable method in the aircraft wing design strategy [2]. After designing airfoil, outer geometry of the wing can be created. Upper and lower skin panel surfaces are designed according to this outer geometry.

Next step in the aircraft wing box design is deciding other structural components in the wing such as spars and ribs as shown in Figure 7.

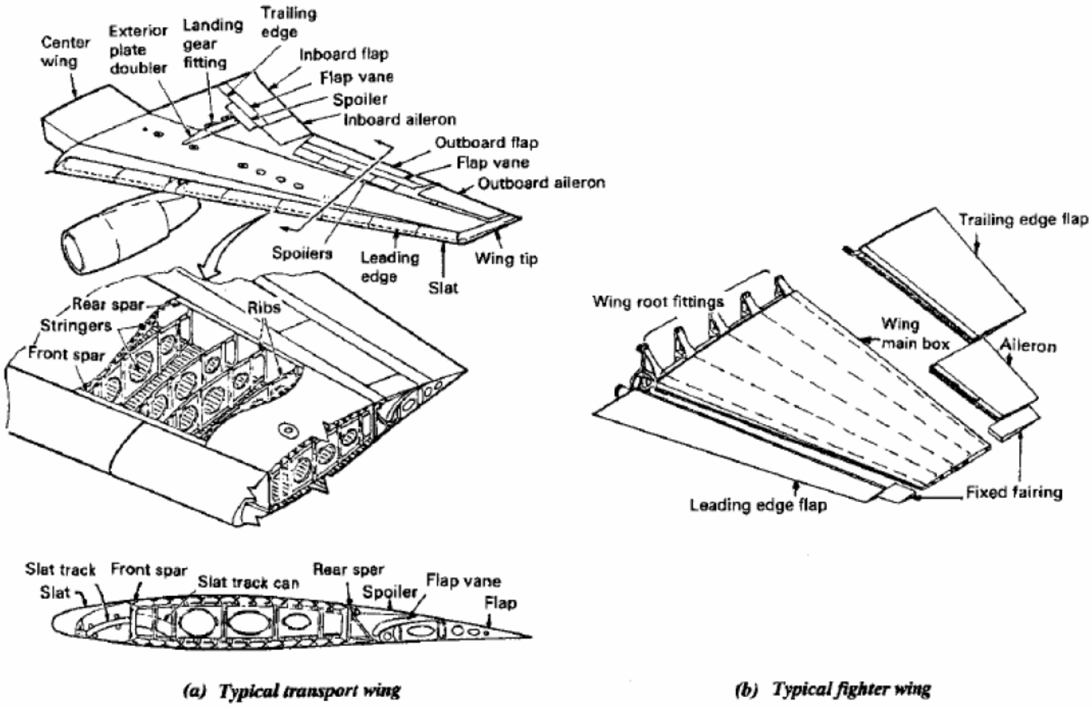


Figure 7. Typical aircraft wing components [3]

In the aircraft wing box design, it will be noted that any wing requires longitudinal (lengthwise with the wing) members to withstand the bending moment which are greatest during flights. This is significantly important for the cantilever wings, which are normally employed for high-performance aircraft. Some of the aircraft have external struts for wing bracing. These types of aircrafts do not require those structures needed for the cantilever wing. The aircraft wing employed in this optimization study does not include any type external bracing since it as a cantilever wing type in the present study [3].

The structural components whose job is to withstand bending moments are named as spar. Spar components continue from wing root to wing tip in lengthwise direction. Number of spars in the wing box is decided in the design stage. With the increasing number of spar, in other words decreasing distance between spars, overall strength of the structure increases as expected. On the other hand, total weight of the wing increases with the increase in the number of spars. There is a tradeoff over this issue and optimum number of spar should be selected in design stage.

Moreover, since the bending moment on the wing root is greater than at the wing tip, spar thickness should be increased from wing tip to wing root as shown in Figure 8. Also some common types of spar configurations are shown in Figure 9.

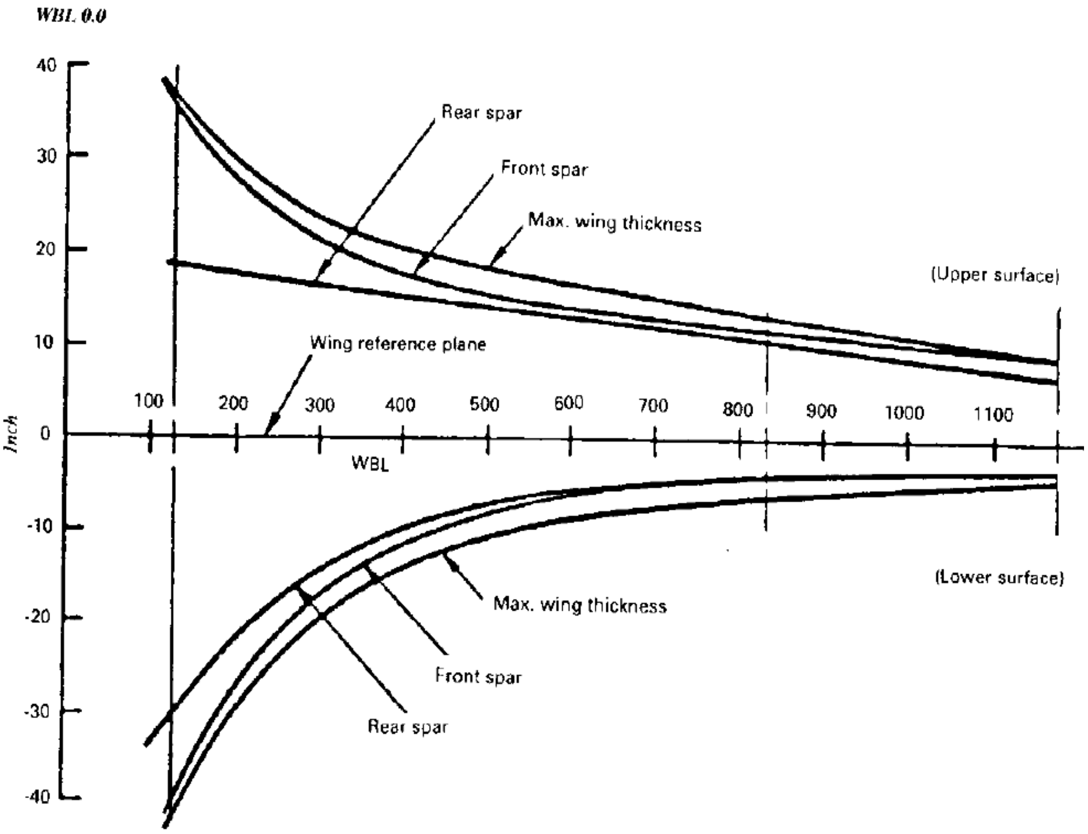


Figure 8. Preliminary view of spars [3]

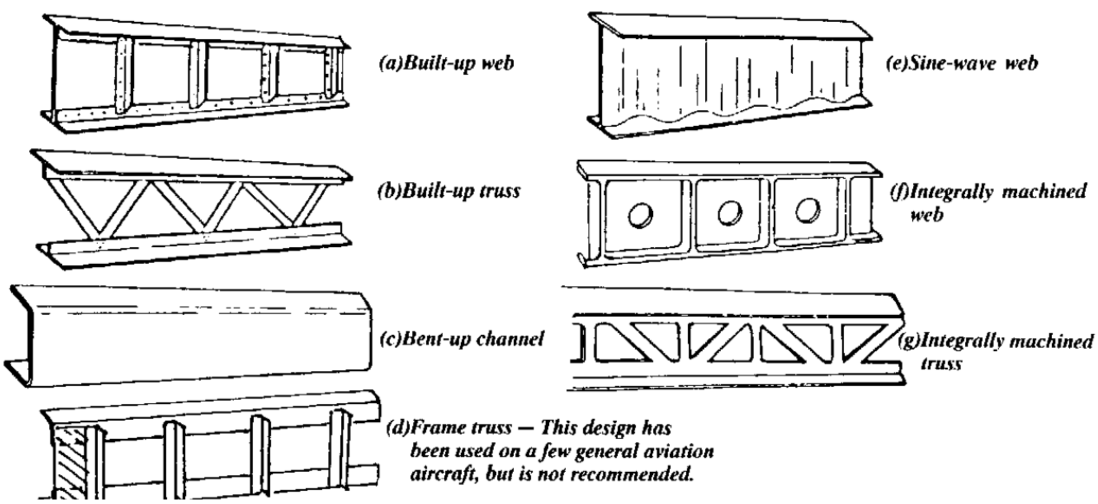


Figure 9. Typical spar configurations [3]

For aerodynamic concerns, the outer wing contour must be maintained without appreciable distortion. For this reason, aircraft wing ribs are used to hold skin panels in the desired contour. Ribs are primarily used for increasing shear strength of the aircraft wing box designs. Number of ribs or rib spacing should also be defined in early stage of the design procedure. Similar to the spars, the increase in the number of ribs increases the overall strength together with the total weight. Therefore, designers should decide an optimized solution. This phenomenon is illustrated in Figure 10 [3].

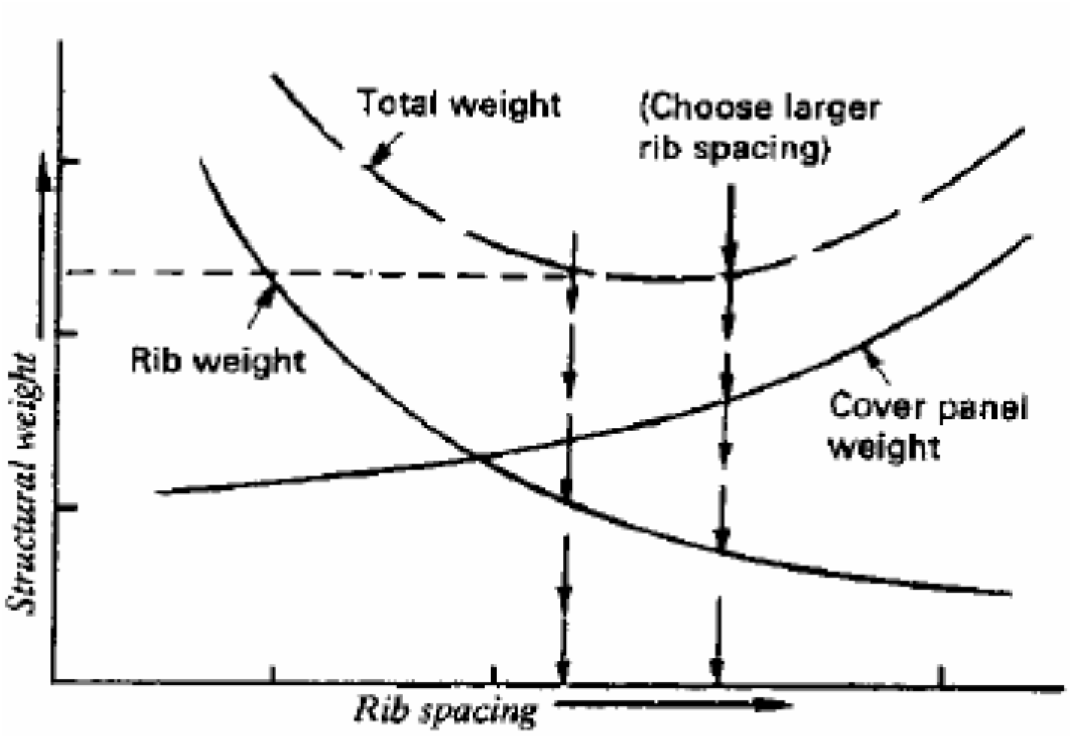


Figure 10. Determination of rib spacing by structural weight comparison [3]

A similar research related to present study was conducted by Çakır [4]. It focuses on structural optimization of a trainer type aircraft wing with the help of genetic algorithm. Although optimization method is different from present study, the wing box design procedure was considered for this study. It gives details about design of almost all aircraft wing box components including the determination of the number of spars, rib spacing etc. It is also important as the initial sizing of the wing box components should be performed according applied load types. Also, it's exemplarily wing box design is introduced at the end of study with the dimensional parameters and thicknesses.

Another similar study is performed by Noman et. al. [5] as named as “Design Analysis of Aircraft Wing Rib”. It focuses on especially aircraft wing rib design with the metallic material of Aluminum 2024-T3. Standard type of aircraft metallic wing rib is considered in this study and optimized. Mainly topological optimization is applied on the design according to load paths on the component. Material cost comparison is performed at the end of the study. Since, weight impacts both material and running costs of the aircraft, those kind of weight optimization based studies are very important in the aviation industry. Similarly, Guo [6] focus on optimization process of an aerobatic aircraft wing structure. Guo [6] also aims at presenting an investigation into a minimum weight optimal design of an aerobatic aircraft wing structure. The wing box between the front and rear spar of a wing is assumed as primary structure and the principal load carrier of the wing in this research. Guo [6] mainly uses aeroelastic optimization method in this research. Another similar study is performed by Neufeld et. al. [7] as named as “Aircraft wing box optimization considering uncertainty in surrogate models”. Neufeld et. al. [7] is also focus on optimization procedure on aircraft wing design. Wing box design used in the research consists of 19 ribs, the front and rear spar, six stringers, and the upper and lower skin. This research aims to presents uncertainty in the conceptual design of the wing box of a generic light jet. Neufeld et. al. [7] uses reliability based design optimization (RBDO) in order to ensure having a feasible design solution.

2.3 Weight Optimization Concept

Weight optimization is the second major part of this study. Since minimizing the weight is the generally primary concern, literature content about weight optimization and weight reduction are very rich. Several optimization method and techniques applied for this purpose can be seen.

About the first study to be investigated here the research performed by Conti et. al. [8]. The study focuses on obtaining optimal stacking sequence and thickness by using finite element method. It takes primarily symmetric and balanced composite structural component design and applies optimization method on it. Sample boundary conditions, loading conditions are applied on the sample initial design and optimal and more efficient structural optimized design is obtained at the end of study by using finite element analysis. This study reveals the importance of weight optimization and weight reduction of structural components.

Similarly, Querin et. al. [9] focuses on weight reduction methods. In this study, layered structural components are optimized by using finite element method and the weight of the structural components is optimized. Main difference from Conti et. al [8] is that it uses topology optimization methods. Topology optimization is a useful method that is used to optimize weight with the given design space. This method also affects the structural shape and design according to load paths on the components as shown in the Figure 11. Although there are several examples of the topology optimization of the structural components in the literature, there are some basic disfavor aspects of it such as manufacturability problem or design based problems in laminated structures [10].

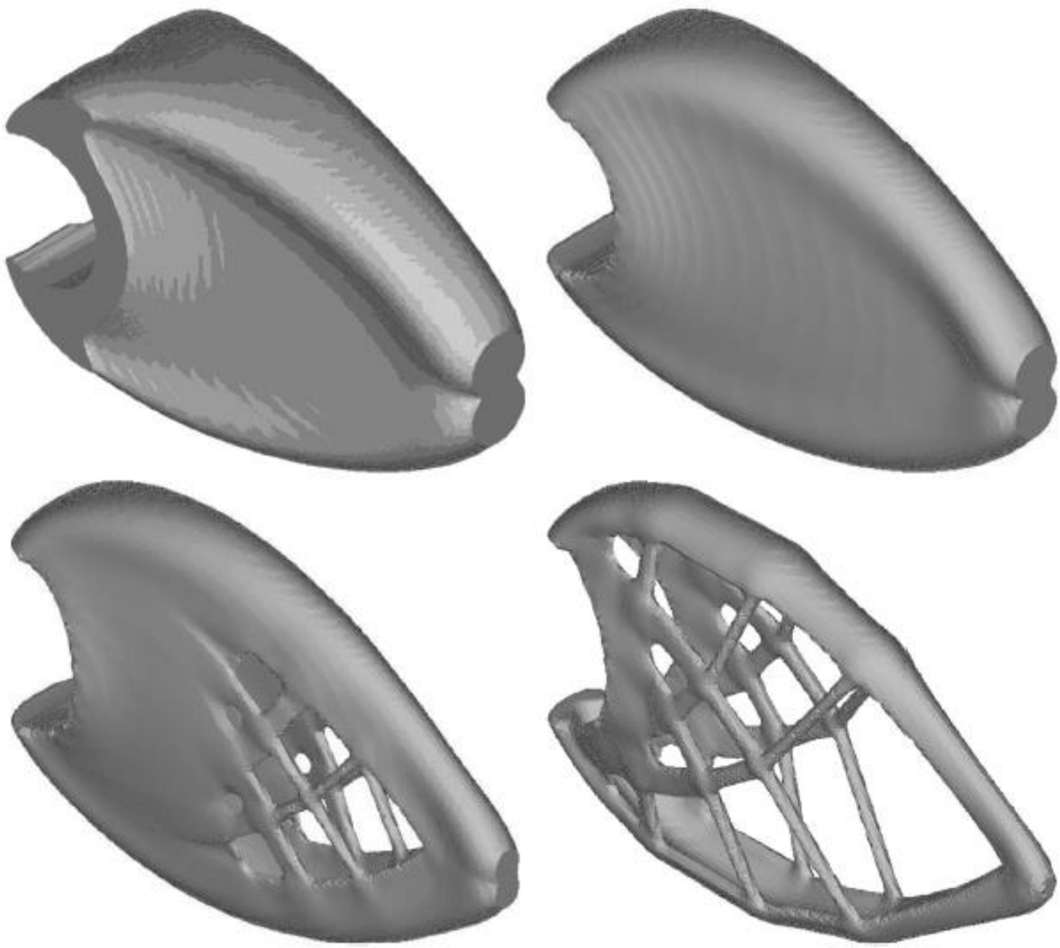


Figure 11. Isometric views of the topology optimization stages [10]

According to the literature, renewable energy industries, specifically wind turbine industry, also focus on weight optimization methods. As weight is a very important efficiency parameter of the wind turbines energy processes. Chen et. al.[11] and Albanesi et. al. [12] performed researched to reach a weight optimized wind turbine and stacking optimized layered structural design with the help of finite element method. They both aimed to reduce structural design weight of the initial designs. In Albanesi et. al. [12], some crucial and key analysis types were defined such as maximum deformation, maximum stress or natural frequency etc. Research concludes that it is possible to obtain up to %15 weight reduction with the optimized design.

Satish et. al. [13] focuses on the weight optimization of a selected aircraft component under sample load case. Also aircraft wing rib is chosen as an exemplarily structural part. Although selected aircraft component is same, there is a major difference since the aircraft wing rib material is metallic. Moreover, its sample structural design is very basic model as shown in Figure 12. Satish et. al. [13] basically aims to optimize number of hole on the component that is used to prevent buckling failure, thickness and cross section of the component. Another main difference is that using ANSYS software as a finite element analysis program. In the present study, Hypersizer software is used in the optimization stage.

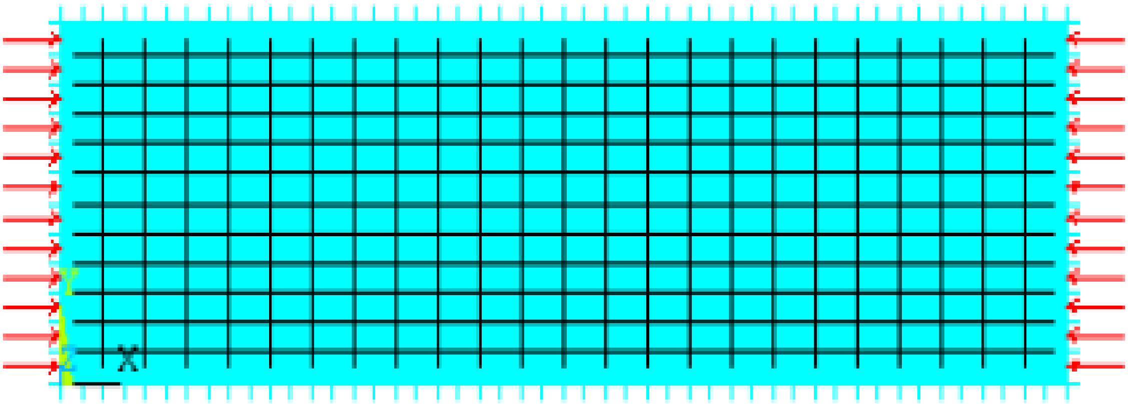


Figure 12. FE model of a sample rib [13]

2.4 Summary

This section of the present work mentioned conducted researches about aircraft wing design and weight optimization methodology. The subject of the present thesis comprises these major fields, and researches mentioned in this chapter are used as reference. Main aim of the present work is to obtain weight efficient aircraft composite wing rib design. To accomplish this goal, an initial aircraft wing design is needed to obtain load distribution on the rib components. Aircraft wing design concept is investigated in the literature and a standard aircraft wing box design is created by following similar research in the literature. After having aircraft wing box design, weight optimization methodology is investigated in the literature. There are several weight optimization methods in the conducted researches such as genetic algorithm, topology optimization etc. After investigating optimization studies, finite element method is selected for structural weight optimization in the present study.

3 THEORETICAL BACKGROUND

3.1 Introduction

This chapter serves the theoretical background information lies behind the present work. This theoretical background section starts with the principles of aircraft dynamics. Then, optimization methodology is introduced. Finally, the summary of the chapter is given.

In the first section, theory behind the aircraft dynamics is explained. To be able to apply weight optimization method in this work, an aircraft wing box model should be obtained. It is hard to have a commercial aircrafts’ wing box design due to various reasons; then a wing box model is designed from scratch in this work. From this perspective, basic theoretical aircraft design methodology should be considered to create new wing box design. This theoretical background information about aircraft design explained in this aircraft dynamic section.

Second section in the theoretical background chapter is the optimization methodology. As mentioned before, this work aims to decrease weight in a specified aircraft wing box component, composite aircraft wing rib. Designer should be familiar to optimization methods to apply weight reduction techniques. In the second part of the theoretical background section, this optimization related information is mentioned. Moreover, Hypersizer software is used in this work and its optimization methods used in weight reduction processes explained briefly. Lastly, brief summary of the theoretical background section is given in the summary section.

3.2 Aircraft Dynamics

Flight terminology means developing flying machines which are heavier than air. This issue is the main purpose of the humanity for centuries with various reasons such as military, medical or transportation. A lot of famous scientists, philosophers and thinkers dreamed to make flying possible [14].

Figure 13. Leonardo da Vinci plans for a flying machine, 1490 [14]

In September 18, 1901, Wilbur Wright and his brother Orville Wright conducted an experiment named as “Some Aeronautical Experiments” in Ohio, US and explain their methodology for developing flying machines to the engineers. They present difficulties which obstruct the pathway to success in flying machine construction in three different steps as follows. Those three general classes of problems are still applicable to modern aviation industry [14].

1. Those which relate to the construction of the sustaining wings.
2. Those which relate to the generation and application of the power required to drive the machine through the air.
3. Those relating to the balancing and steering of the machine after it is actually in flight.

First class of problem is related to lift force generation of the airplane needed to stay in the air. Second one is related to engine power of the airplane and last one is about controlling to the airplane in the flight conditions. In 1901, Wright brothers somehow solved those problems with their own methods and achieve to design first airplane in the history [14].

Figure 14. Wright brothers' first flying machine, Orville Wright at the control, passing over Huffman Prairie, October 4, 1905, Washington, D.C [14]

As mentioned previously, those problems are still applicable in the modern aviation industry and engineers still should consider those challenges in their airplane design stages. In this chapter of the present study, brief information about first and third main class of problems Wright brothers declared in 1901 mentioned in separate sections.

3.2.1 Lift Force Generation

Lift is the force means directly resist to the weight of the aircraft and makes possible to stay in the air airplanes. This force is generated mainly at two wings of the airplane. Wings are structural parts of the airplanes that are responsible to generate lift force to hold airplanes in the air. It is generated by the interaction of the solid wing part and fluid air. Fluid air flows with different speed at the lower side and upper side of the wing panels and this generates pressure difference between upper and lower panels of the wing box as shown in Figure 15. This pressure difference creates a mechanical force named as lift force.

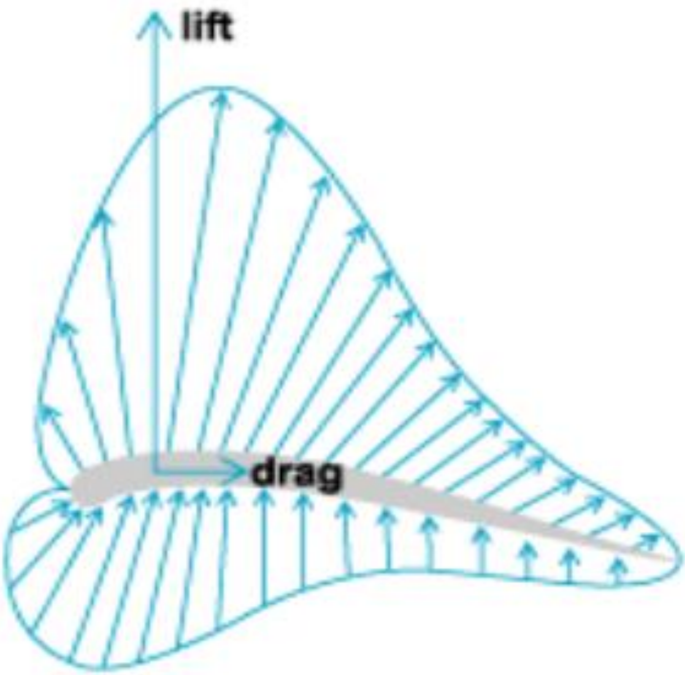


Figure 15. Example pressure distributions on an airplane wing

This entire phenomenon obeys a rule, called Bernoulli principle. Bernoulli principle is generally a set of variations on an equation that explains the relationship between static pressure, dynamic pressure, and manometric pressure. Common known forms of the theorem are shown in equation (2) and equation (3) [15].

$$\mathrm{P} + \frac{1}{2}\rho \mathrm{V}^2 + \mathrm{g}\rho \mathrm{y} = \mathrm{Constant} \tag{2}$$

In other terms;

$$\mathrm{P}_1 + \frac{1}{2}\rho {\mathrm{V}_1}^2 + \mathrm{g}\rho \mathrm{y}_1 = \mathrm{P}_2 + \frac{1}{2}\rho {\mathrm{V}_2}^2 + \mathrm{g}\rho \mathrm{y}_2 \tag{3}$$

Where P is static pressure, ρ is density of the fluid, v is speed of the fluid, y is height above predefined datum and g is the gravitational acceleration. It makes some restrictive assumptions like steady flow of the fluid, no viscosity of the flow, incompressible fluid etc. [15].

When this principle applied to the streamlines that goes from upper and lower side of the airfoil, one of them goes faster with respect to other one. So, a pressure difference is occurred and lift force is generated as shown in Figure 16.

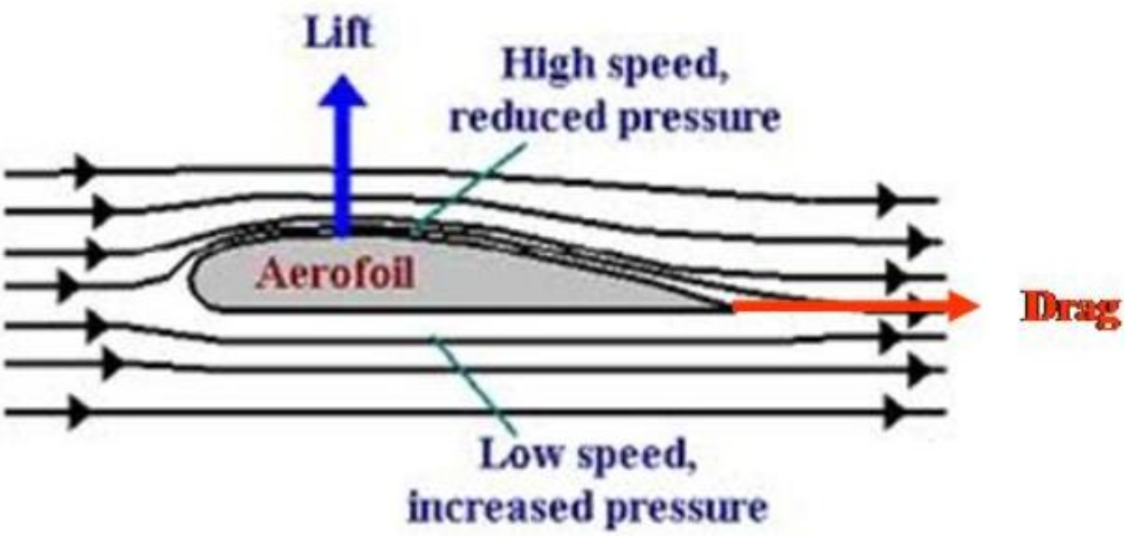


Figure 16. Stream lines around airfoil and lift force generation

When lift force concept is considered, it is easy to see that airfoil geometry has a high impact on the lift force generation. Then, design of airfoil geometry and type are considered. Main geometrical parameters and general overview of the standard airfoil type is shown in Figure 17.

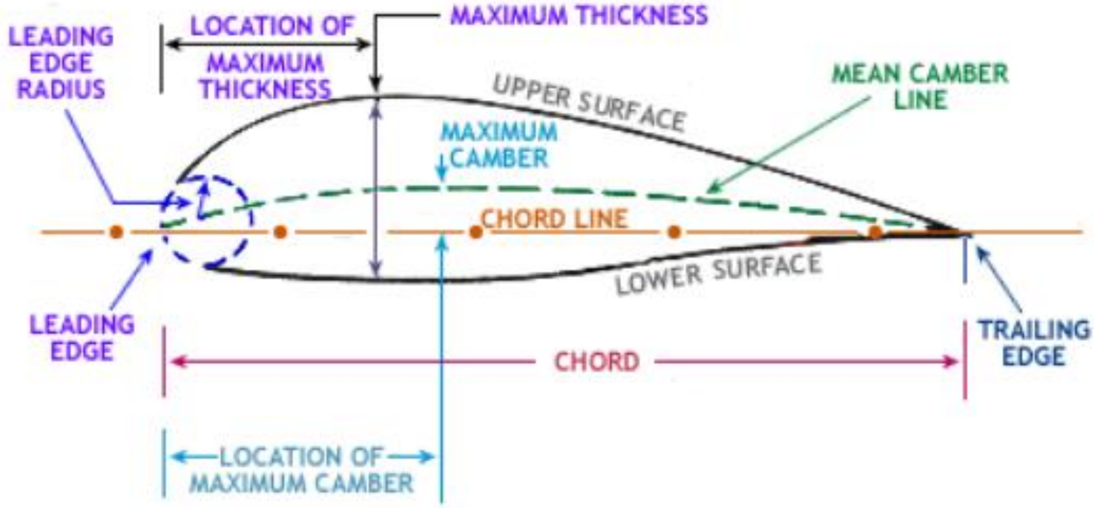


Figure 17. Overview of standard airfoil geometry

It is clear that defining all this parameters and geometry of the airfoil is destitute to large scale expensive experiments and surely solving tons of explicit mathematical functions. To avoid this difficulty, designer may use standard airfoil types according to design requirements. A variety of airfoil families are ready to use for the designers as briefly shown in Figure 6.

In today’s modern technology world, choosing airfoil type from catalogs considering some factors is an old-fashion method. Modern engineering applications are based on inverse computational solutions for desired pressure or velocity distribution on the airfoil. It is easy than before with increasing computational power in the technology. However, choosing airfoil type from catalogs is still valid method in the aircraft wing design [2].

3.2.2 Aircraft Wing Box Components

In the aircraft wing design, next step should be configuring the wing after selecting airfoil cross section. There are some common types of structural components such as panels, spars and ribs in the wing box structural design. Their quantity on the wing design, locations and positions should be configured in the wing [16]. To do so, designer should be aware of main duties and profits of components to the structural integrity of the wing. In this section, related background information is summarized.

Main structural components of the standard aircraft wing box design are shown in Figure 2. Design contains upper and lower skin panels, spars and ribs. All those structural components have their own responsibilities and duties in the wing box design. Each of them is responsible to resist for different types of stresses according to the flight condition of the aircraft.

Design in Figure 2 represents a standard wing box with two C type spars, ribs including closure ones and upper and upper and lower skin panels.

Skin panels are mainly responsible for the aerodynamic forces. Aerodynamic pressure and force act on the upper and lower skin panels first and then split into other structural components of the aircraft wing box design. So, panels are mainly responsible for aerodynamic consistency of the outer shell of the wing box design [2].

Aircraft wing can be thought as a simple cantilever beam in general. From this perspective, it is seen that compression loads act upper shell and tension loads act lower shell with standard upward load case as summarized in Figure 18 [2].

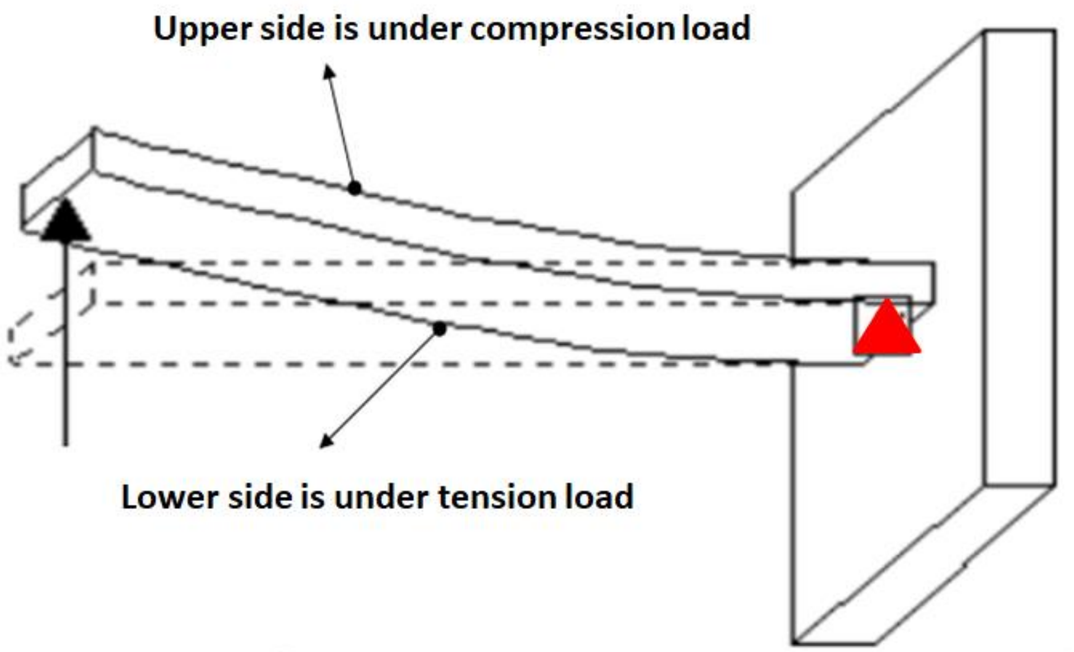


Figure 18. Cantilever beam deflection

Other main structural part is spar. Spars are generally placed in the wing box design to make stiffer it under the bending load type. It makes possible to resist to the bending load of the wing. Its stress distribution is, as expected, increasing from wing tip to wing root since it acts like exactly cantilever beam. With this reason, root side of the spars can be design thicker than the other sides of it. It can be C types or I types as shown in Figure 19 [2].

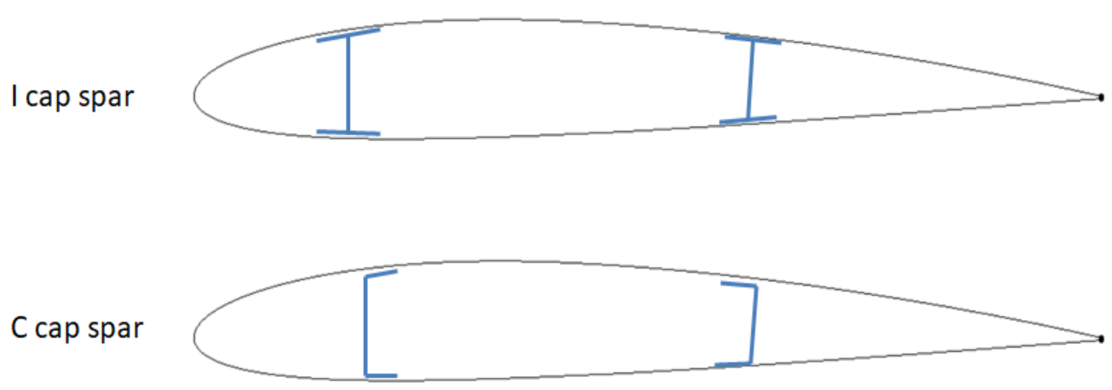


Figure 19. Spar cap types illustration

The other structural components are ribs. It is generally placed between spars orthogonally. It lays in chord direction on the wing and stands against shear dominant loads on the wings. As in case of spar, their quantity and spacing should be decided according to load cases that size the aircraft carefully. Number of ribs and spars in the wing and rib & spar spacing of the wing box design have great impact on the load distribution of the wing [2].

3.3 Optimization Methodology

Structural optimization words basically mean removing inefficient material sections from designed parts by following some predefined objectives and aim to evolve design to optimum one. This is the basic definition of structural optimization process. Through this processes, there might be several design constraints such as stress or strain level, deflection level, buckling condition, stiffness or frequency. These constraints shape main objective functions of the optimization processes and criteria for removing materials from the structural design [17].

Traditional structural optimization process follows a parametric path like defining shape or structure of the design by using a set of design parameters. After that, it defines the relations between those design parameters with the objective functions like stress level, deflection level, buckling condition, stiffness or frequency etc. Then, it tries to minimize those objective functions by changing those design parameters variables repetitively [17].

Most applicable and beneficial optimization is called as material removal based on stress level. At the beginning of this type optimization method, large enough material to cover of the assumed final design is transferred into the finite element model environment. It is applied to good enough fine mesh. After applying enough fine mesh in the computer aided engineering environment, loads and boundary conditions are applied and a stress analysis is carried out using same computer software. It helps to designer in deciding material sections that is stressed and unstressed. Unstressed regions or relatively small stress sections are defined and they are applied to rejection. Using this rejection, called rejection criterion, such inefficiently used material may be eliminated. Removal of the material from structural design can be seen as deleting small mesh elements from the finite element model. [17].

Stress value of each element can be computed very differently. One of them is von Mises stress which is most frequently used criteria for isotropic materials for this purpose. For plane stress problems, the von Mises stress can be defined as shown in equation (4).

$$\sigma^{vm} = \sqrt{\sigma_x{}^2 + \sigma_y{}^2 - \sigma_x\sigma_y + 3\tau_{xy}{}^2} \tag{4}$$

Where σ_x and σ_y are normal stresses in x and y directions, respectively, and τ_{xy} is the shear stress.

By removing unused or inefficient material sections from the structural design, very big amount of mass saving can be achieved. Figure 20 and Figure 21 represent applications of this procedure and initial and final structural designs after applying stress based structural optimization method.

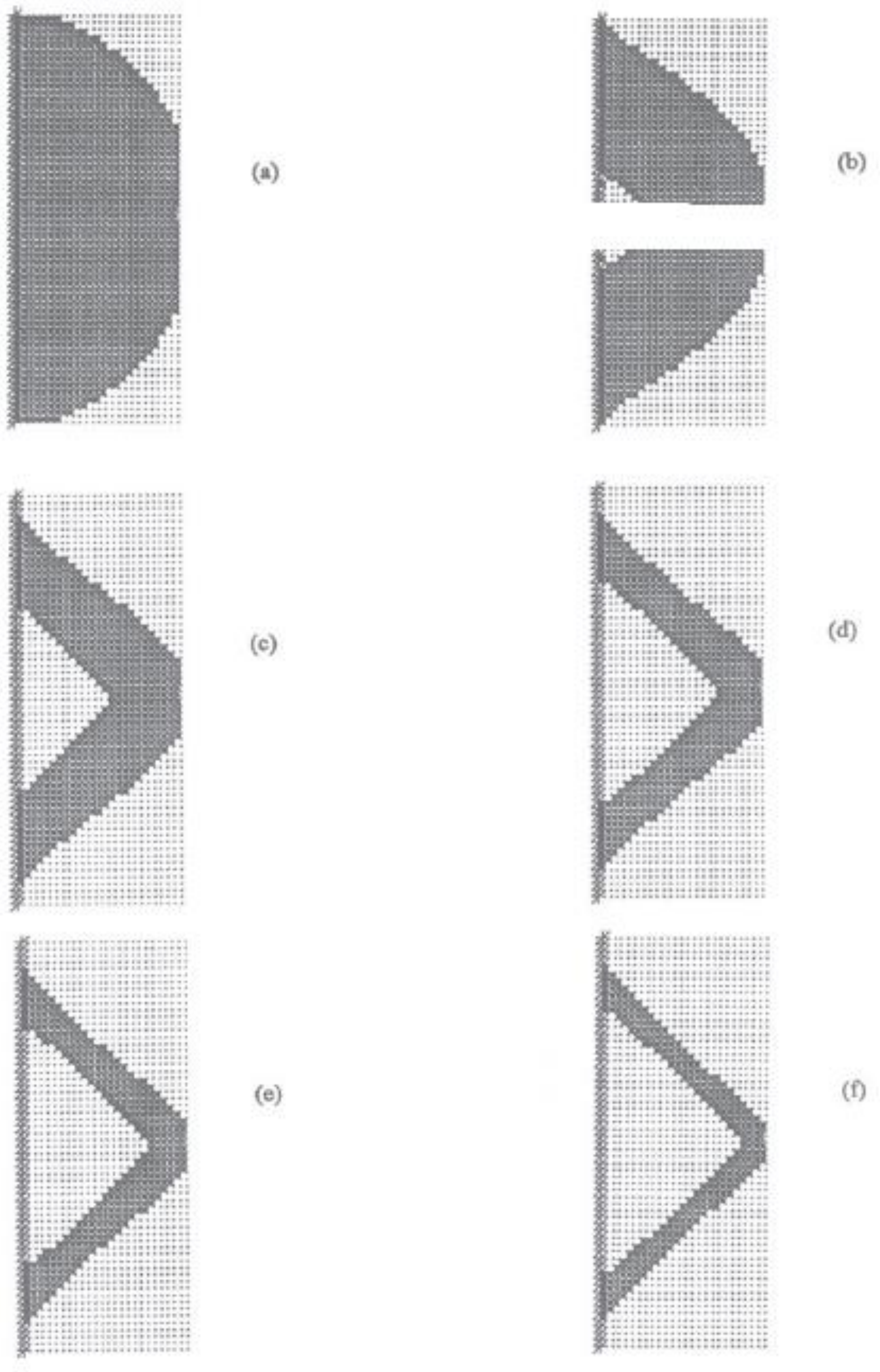


Figure 20. Structural optimization stages of an example design [17]

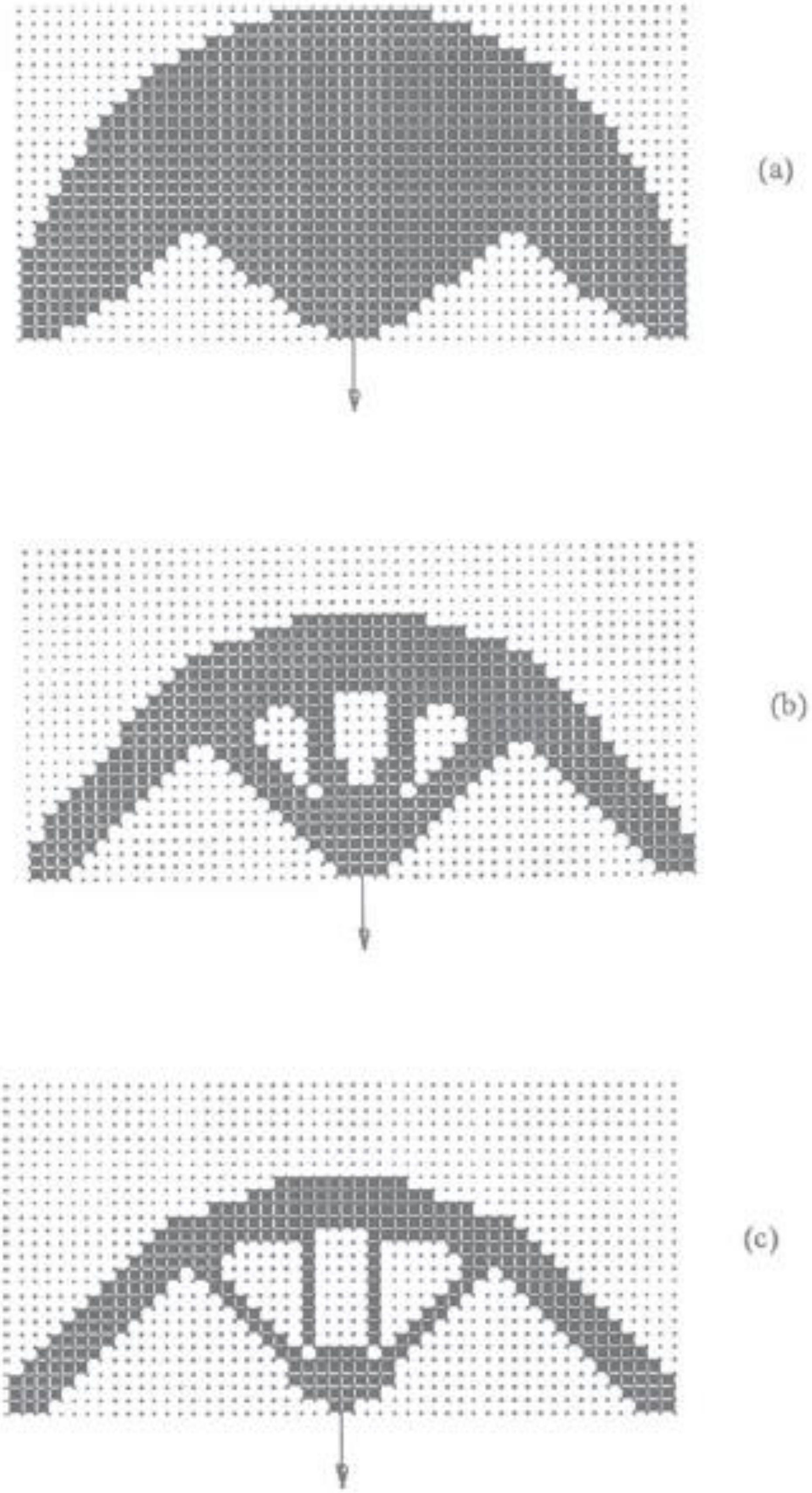


Figure 21. Structural optimization of Michell structure with fixed supports [17]

As mentioned previously, Hypersizer computer software is used in this work as an optimization tool. It has its own optimization methodology as follows.

HyperSizer is software that mathematically coupled with MSC/NASTRAN solver software to provide an integrated solution for quick and accurate design optimization specially thickness optimization. Biggest difference of it from the traditional structural optimization tools is that it updates stiffness of the structure after all iterative cycle on the structure. Although it works for both metallic and composite structures, it is very useful and beneficial for composites because it provides us optimized stacking sequence of the composite layered structures by considering a lot of user defined constraints and manufacturability aspects [18].

It is not FEA software but it couples with some of the commercial FEA program and takes grid point deflection and element forces from FEA program. After that, it applies its optimization methods to the component and obtains producible new optimized design solutions. This procedure is summarized in Figure 22. It calculates margin of safeties in all predefined local regions and marks over safe design areas on the structural components.

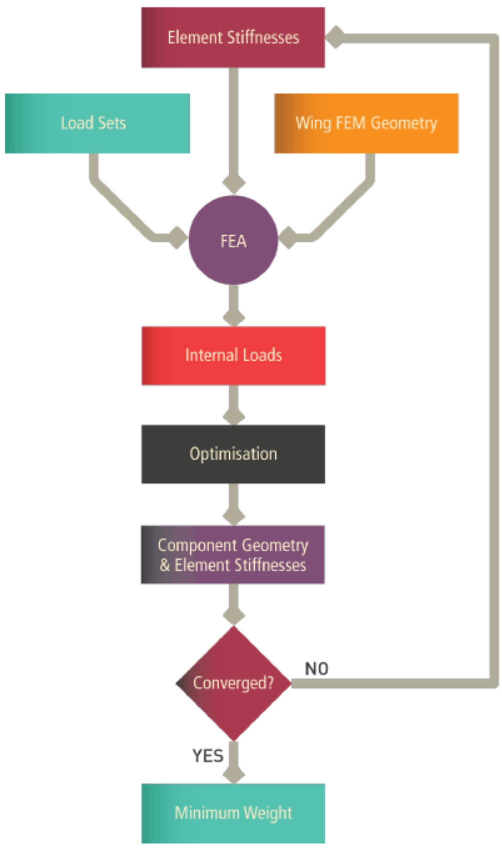


Figure 22. Hypersizer work flow summary [19]

Since Hypersizer software provides us a detailed analysis including thickness optimization, stacking optimization, margin of safety calculations etc. it is very useful in the aviation industry. It also gives very quick review about the structural design and its initial geometrical parameters in the early stage of the design process [18].

3.4 Summary

Both designing aircraft wing box from scratch and structural optimization and mass decreasing operations on it are very massive duty and it contains several details in it. In this chapter, firstly, critical key points of the aircraft wing box design are mentioned. It is explained with the aviation structural design history from Wright brothers. After that, structural optimization is defined and important or noticeable details about the structural optimization and mass saving methods are demonstrated.

4 AIRCRAFT WING BOX DESIGN

4.1 Introduction

The workflow of the entire process is presented in Figure 23. Red rectangle represents the topic explained in this chapter. First stage of the work is wing box design of the aircraft. The design process includes airfoil selection, choosing geometrical values used in the design and the selection of materials. Common aircraft wing box design rules and similar works that are done in the literature are followed in this process. The aircraft wing box design established a ground for next steps and optimization is carried out with the results of this stage.

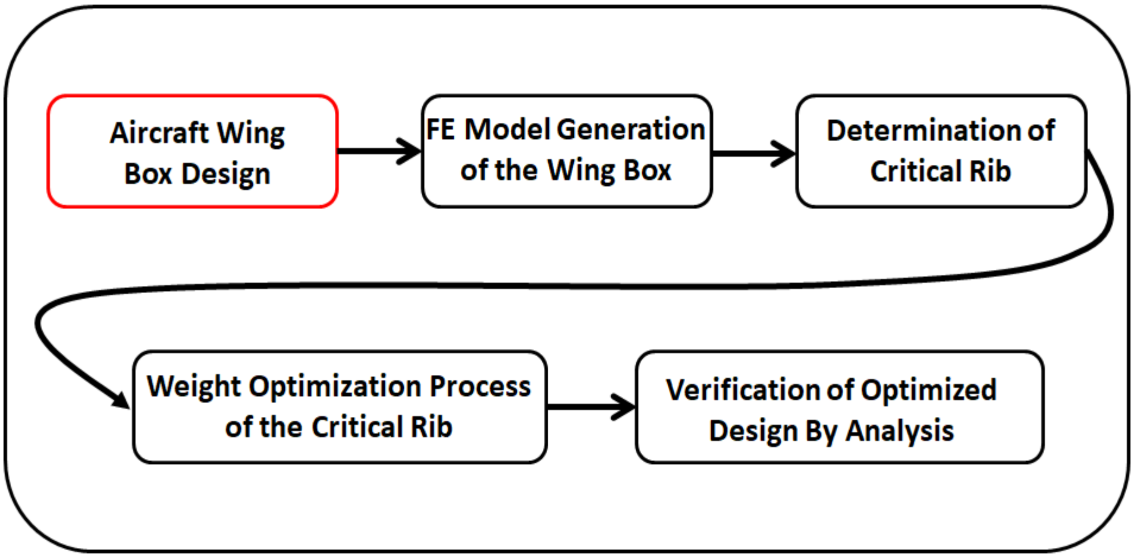


Figure 23. Flow chart of the work

4.2 Airfoil Selection

As mentioned in the Chapter 3, theoretical background chapter, there are some ready to use, open source airfoil families. For this work, NACA airfoil family is chosen and NACA 2412 Airfoil is implemented into the wing box design. Point cloud of the airfoil surface was already given as open source in the internet and is given in Table 1.

Table 1. NACA 2412 airfoil points on surface

Point Number	X Coordinate	Y Coordinate
1	1.00008	0.00126
2	0.99856	0.00158
3	0.99398	0.00252
4	0.98639	0.00409
5	0.97583	0.00623
6	0.96234	0.00892
7	0.94603	0.01211
8	0.92697	0.01574
9	0.90529	0.01975
10	0.88110	0.02408
11	0.85457	0.02865
12	0.82583	0.03340
13	0.79507	0.03826
14	0.76247	0.04315
15	0.72823	0.04800
16	0.69255	0.05274
17	0.65567	0.05730
18	0.61779	0.06162
19	0.57916	0.06561
20	0.54001	0.06922
21	0.50059	0.07238
22	0.46114	0.07503
23	0.42192	0.07712
24	0.38303	0.07857
25	0.34468	0.07920
26	0.30729	0.07894
27	0.27111	0.07780
28	0.23637	0.07579
29	0.20331	0.07295
30	0.17215	0.06931
31	0.14309	0.06494
32	0.11631	0.05992
33	0.09200	0.05433
34	0.07029	0.04826
35	0.05132	0.04181
36	0.03521	0.03508
37	0.02205	0.02815
38	0.01191	0.02112
39	0.00483	0.01405
40	0.00086	0.00700
41	0.00000	0.00000
42	0.00222	-0.00669
43	0.00748	-0.01283
44	0.01572	-0.01840
45	0.02689	-0.02341
46	0.04091	-0.02783
47	0.05767	-0.03165
48	0.07707	-0.03488
49	0.09899	-0.03751
50	0.12328	-0.03955
51	0.14981	-0.04101
52	0.17840	-0.04193
53	0.20890	-0.04235
54	0.24113	-0.04229
55	0.27490	-0.04183
56	0.31003	-0.04103
57	0.34630	-0.03994
58	0.38352	-0.03864
59	0.42164	-0.03717
60	0.46040	-0.03544
61	0.49941	-0.03349
62	0.53845	-0.03137
63	0.57728	-0.02914
64	0.61566	-0.02683
65	0.65335	-0.02450
66	0.69013	-0.02217
67	0.72576	-0.01988
68	0.76003	-0.01765
69	0.79272	-0.01550
70	0.82362	-0.01345
71	0.85254	-0.01151
72	0.87930	-0.00970
73	0.90373	-0.00803
74	0.92567	-0.00652
75	0.94498	-0.00517
76	0.96154	-0.00401
77	0.97523	-0.00304
78	0.98598	-0.00227
79	0.99371	-0.00171
80	0.99836	-0.00137
81	0.99992	-0.00126

It has 81 discrete points on the airfoil outer surface. While connecting those 81 points together by using spline, outer surface of standard NACA 2412 airfoil is obtained. Points are given with cosine spacing and close trailing edge options selected on the source where those points are taken. Those points are introduced to using Catia (version 5) software and a spline is generated through these points to obtain the airfoil form (Figure 24).

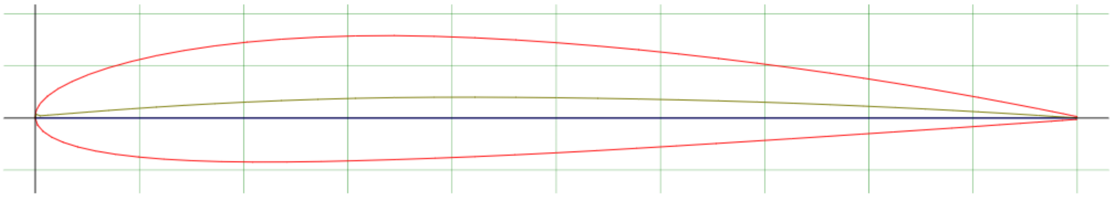


Figure 24. NACA 2412 airfoil outer shell

The outer shell of the NACA 2412 airfoil has unit chord length, thus, it is scaled according to desired chord length in the wing tip and root.

Wing box is defined between from %7 to %62 distances from leading edge of the airfoil between front and rear spar as demonstrated in Figure 25. Moreover, aircraft wing twist angle is not considered in this airfoil selection process for the simplicity.

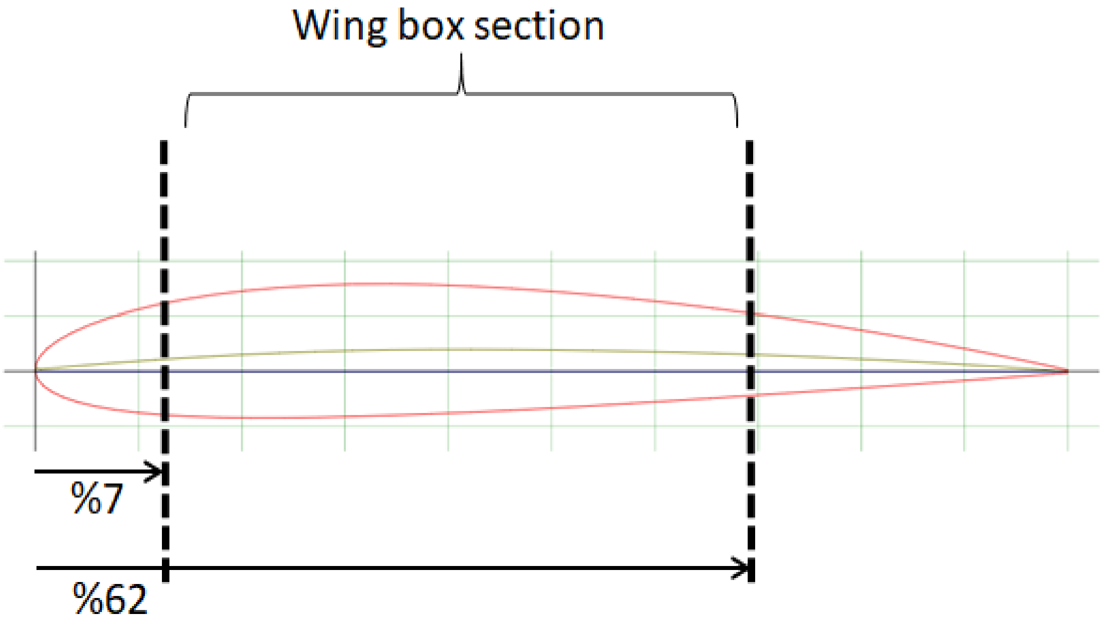


Figure 25. Wing box region of the airfoil

4.3 Geometry Description

The dimensions of the aircraft wing box were determined based on the literature explained in Chapter 2. Çakır [4] contains aircraft wing box design stage for its purpose of use. Most of the rough dimensions as well as number of ribs and spar of the initial model of present work are taken from that study. The dimensions of the generated model in Catia environment is shown in Figure 26. Figure 27 shows the components in the wing box assembly and their configurations. Each rib is modeled with 255 mm spacing in the wing box design. All components initial thicknesses (before optimization process) are taken from Çakır [4] and given in Table 2.

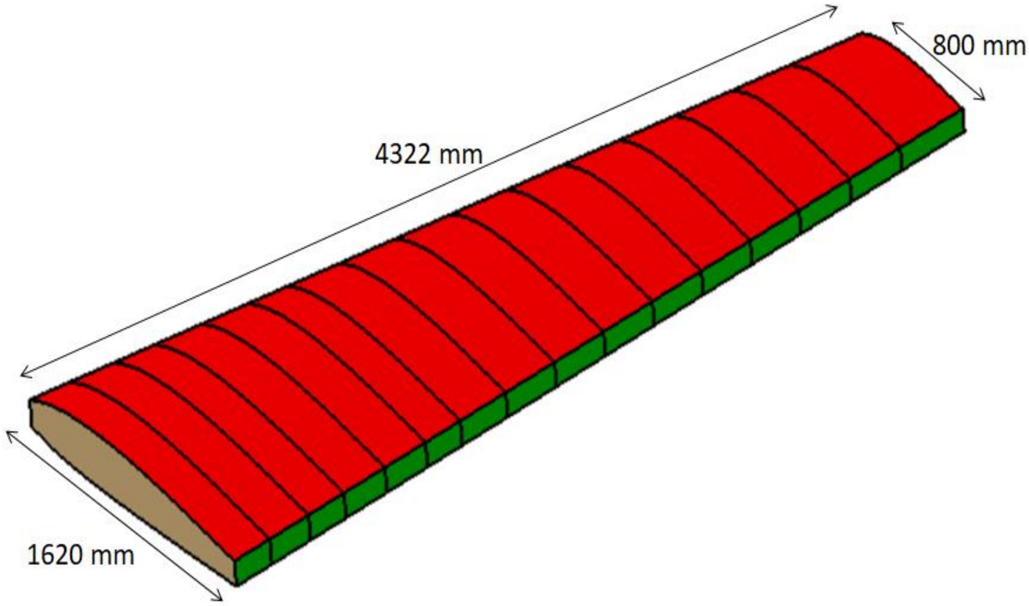


Figure 26. Dimensions of the wing box design

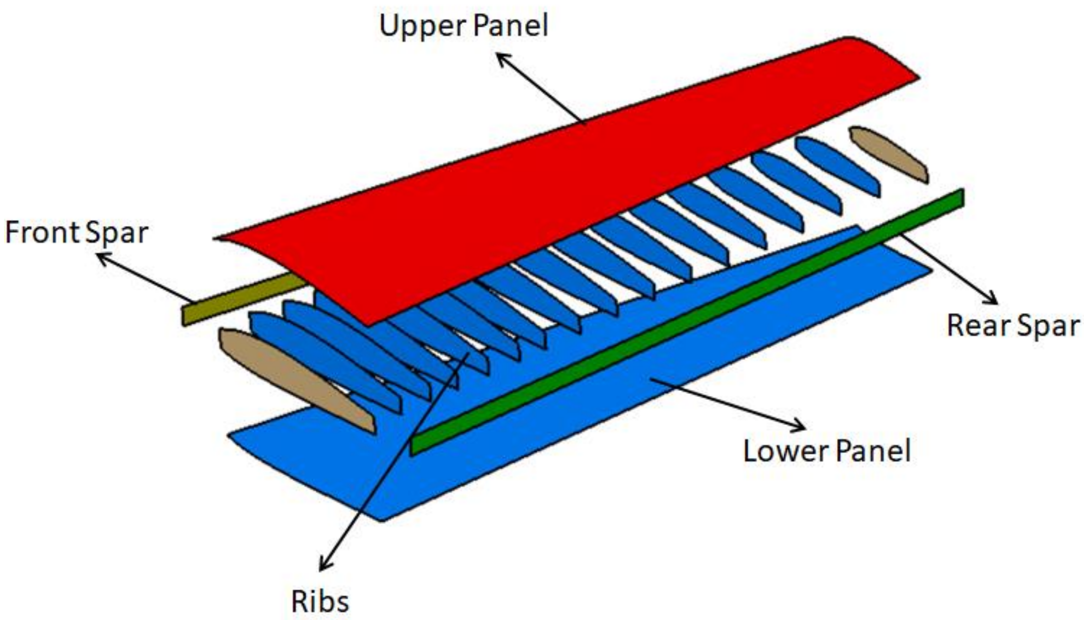


Figure 27. Exploded view of the wing box assembly

Table 2. Wing box components thickness values

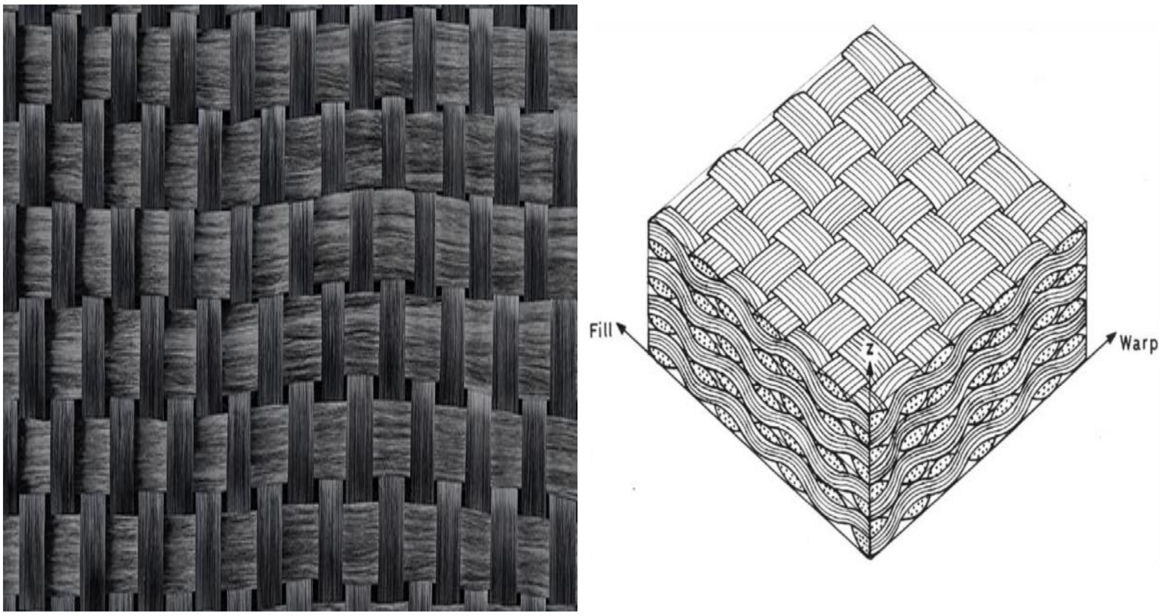
Wing Box Component	Thickness (mm)
Upper Panel	2.60
Lower Panel	2.40
Front Spar	3.10
Rear Spar	2.70
Ribs	2.18

4.4 Material Description

All components except ribs are modeled with the metallic material. In the common upward load case definition, it is easily seen that upper panel faces with compression load and lower panel faces with tension load. So, material selection should be performed carefully to assure fairly high tensile strength to density ratio combined with good fatigue property. Aluminum and titanium alloys are very stiff in both under compression and tension load. Aluminum alloys such as 2024-T3 and the newer types such as 7475-T7351 are excellent candidates along with most of the titanium alloys such as Ti-6A1-4V [3]. Since, material costs are another big issue in the material selection, 7475-T7351 aluminum alloy is selected as metallic material for the wing box design.

Only ribs are modeled with the composite material in the aircraft wing box design. In aerospace applications, carbon fiber reinforced polymers (CFRP) are commonly used materials. Hexcel 8552 epoxy resin is one of the common matrix material used in these applications since its good impact and damage tolerance behavior. In present study, CFRP composite ribs are designed from the prepregs consist of Hexcel 8552 epoxy resin and AS4 family carbon fibers. Plain weave fabric prepreg is used in the design. Although rib components web sections are generally encountered with the shear dominant loading type, it is possible to face with the bending or axial loadings according to rarely encountered load case definitions of the aircraft. Unidirectional (UD) CFRP are not considered as an option in the design with this reason. Moreover, unit weight of the material used in aerospace industry is important. Low gsm composite materials are better with this reason. 193 gsm fabrics are commonly used type of the composite materials and it is decided to use in aircraft composite wing rib design of present work. The structure of the composite material is shown in

Figure 28 [20]. The materials used in the aircraft wing box design are given in Table 3 according to the components.



(a) (b)

Figure 28. A 2D plain weave woven fabric composite (a) lamina (b) laminate

Table 3. Wing box components materials

Wing Box Component	Material Name
Upper Panel	Al 7475 – T7351 Aluminum
Lower Panel	
Front Spar	
Rear Spar	
Ribs	AS4-8552 Plain Weave Woven Fabric Carbon Fiber Composite

4.5 Summary

Aircraft wing box design needed while obtaining load path on the ribs is defined in this section. Firstly, airfoil type used in the aircraft wing box design is decided and explained. Geometry of the wing box is described and main geometrical properties such as number of ribs or number of spars are explained. Dimensions and thicknesses of the aircraft wing box components are given. Lastly, materials used in the design are given and the selection criteria of those materials are explained.

5 FINITE ELEMENT ANALYSIS OF THE WING BOX DESIGN

5.1 Introduction

The workflow of the entire process is presented in Figure 29. Red rectangles represent the topics explained in this chapter. First stage of the work is preparation of finite element model of the wing box design. Then, finite element analysis of the designed aircraft wing box is performed to determine the critical rib to optimize and to obtain load distribution on it. The prepared solid model is transferred to the Hypermesh software for the preprocessing of the finite element model. The main reason why Hypermesh is used as preprocessor is its high performance capabilities in meshing environment. Hypermesh is user friendly and high quality finite element analysis software. It is able to manage with complex models easily and used in aviation and aerospace industry for the analysis of structures as a preprocessor widely. For the post process purposes, MSC Patran 2019 software is used.

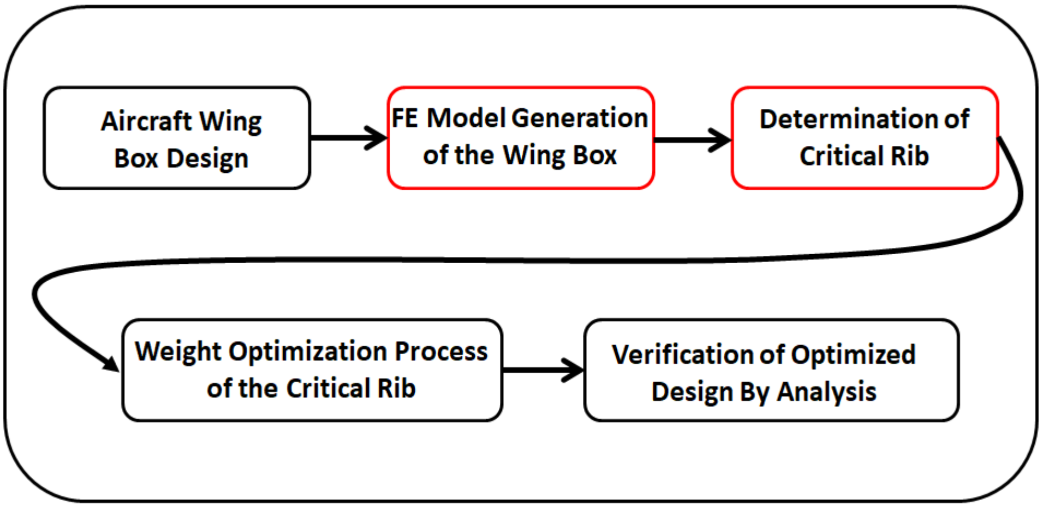
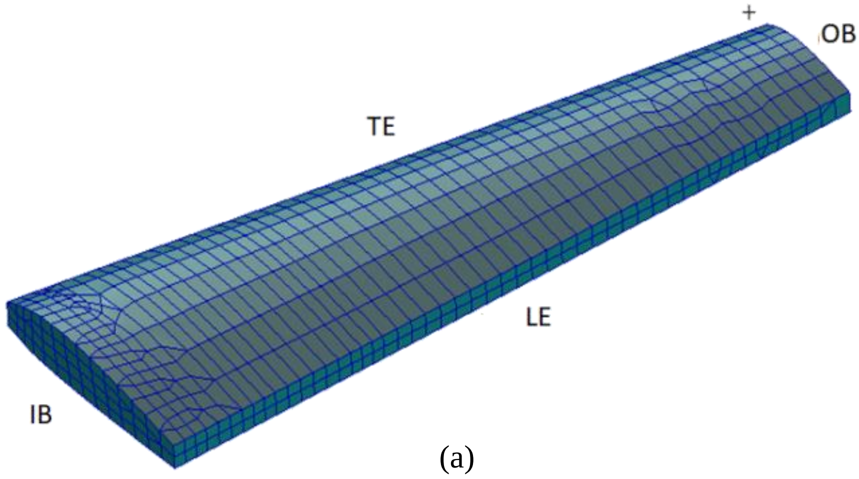


Figure 29. Flow chart of the work

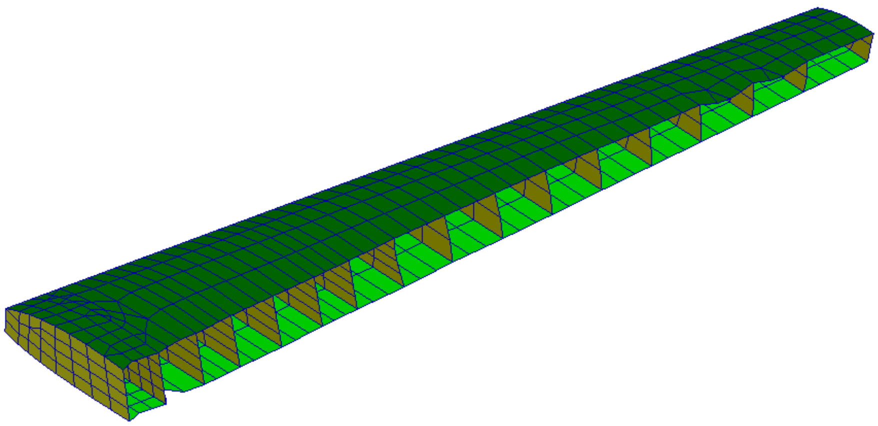
5.2 Finite Element Model Description

Finite element model generated for present work is shown in Figure 30. It is created as a standalone type means connection nodes of the wing box to the rest of the aircraft is assumed as fixed and undeformed.

Finite element model consist of rear spar, front spar, upper panel, lower panel and ribs. Those components are modeled in global finite element model format. Flanges of the components are not modeled since they do not have huge impacts on the load distribution. Moreover, connections between components are modeled by using coincidence node technique with the same reason. Load passes thorough components are flows on those nodes. The finite element model and the mesh structure of the wing box are shown in Figure 30. In Figure 30 a, the global view is shown whereas in Figure 30 b, the crossection from the wing box is given showing the spars, ribs and the panels.



(a)



(b)

Figure 30. FE model of the wing box (a) global; (b) cross-sectional view

Mesh size assigned in the FE model is rather coarse. This FE model is used for only to determine the load path of the ribs. Since mesh size does not contribute to the load distribution, it is kept as coarse in order to spend less computational time. Mesh details are demonstrated in Table 4. Finite element model has totally 7488 degree of freedom.

Table 4. FE model description

Type	Definition	Quantity
CQUAD4	4 node quadrilateral shell element	1352
CTRIA3	3 node triangular shell element	60
GRID	Node	1254
LC	Load case	1
MAT1	Isotropic material card	1
MAT8	Orthotropic material card	1
PCOMP	Property card for composite material	1
PSHELL	Property card for metallic material	4

5.3 Constraints and Boundary Conditions

The standalone finite element model is fixed from spar ending nodes at the root side in 6 degree of freedom. Single point constraints in all 6 directions are defined at the spar nodes connecting wing box to the center fuselage of the aircraft as shown in Figure 31. Patran has a feature called as autospc mode. This feature creates single point constraints at the nodes look like singular in the FE model and fixes those nodes in 6 degree of freedom. This feature is closed at the analysis since fixing nodes with autospc mode of Patran may cause some errors in the model.

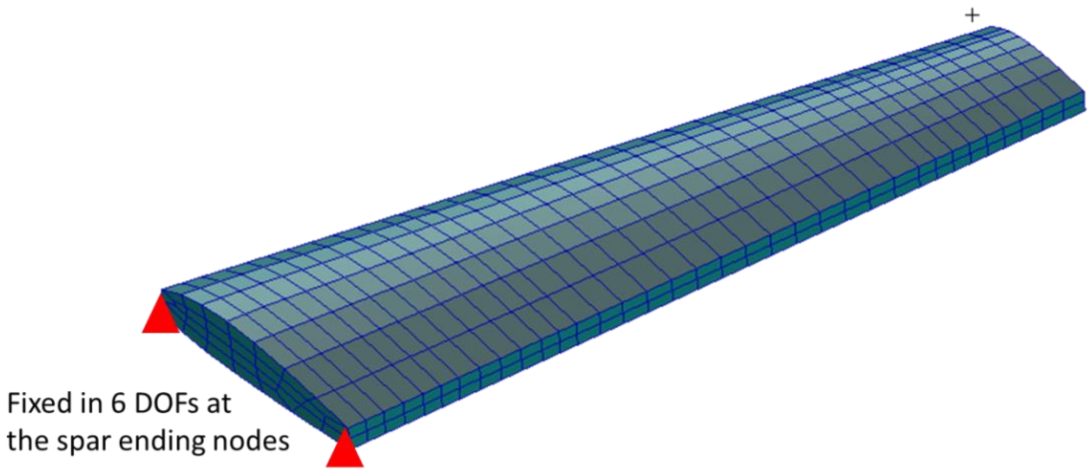


Figure 31. Single point constraints of the FE model

5.4 Load Case Definition

Loads are assigned according to the information given in Çakır [4]. The weight of the designed aircraft to be considered is assumed to be about 3000 kg. Maximum and minimum standard gravitational acceleration limits of the aircraft +7 g and –3.5 g. +7 g is chosen as the design load case since it is able to create most critical scenario [4]. In normal straight-level flight, the wing lift supports only the weight of the airplane. Total lift force is shown in equation (5);

$$F_{lift} = m * a = 3000 * 7 * 9.81 = 206010 \text{ N} \tag{5}$$

This load can be considered as the limit load. The ultimate load is calculated by multiplying the limit load by a safety factor in equation (6). This safety factor is generally taken as 1.5 in aviation industry [4].

$$F_{ultimate} = F_{limit} * SF = 206010 * 1.5 = 309015 \text{ N} \tag{6}$$

This value is the maximum lift force generated in both wings. By dividing this value to 2, the maximum lift force for one wing can be calculated as 154507.5 N.

This load value is the total applied lift force on the finite element model. Lift force is applied to the aerodynamic center in the design of aircraft wings. It can be assumed as quarter of the chord length from leading edge as shown in Figure 32. Application of this distributed load in the finite element model is visualized in Figure 33. The lift force is applied onto finite element model as decreasing linearly from wing root to wing tip due to aerodynamic rules. Constant pressure distribution over wing box lower panel is assumed and force distribution along aerodynamic center is assumed decreasing linearly from wing root to wing tip since area of the aircraft wing box lower panel is decreasing linearly towards the tip. 40 nodes on the aerodynamic nodes are specified and total lift force calculated previously is distributed onto those nodes. Force distribution on those prescribed nodes is shown in Figure 34. Curve in Figure 34 is obtained by considering area ratio of all sections of the aircraft wing box design.

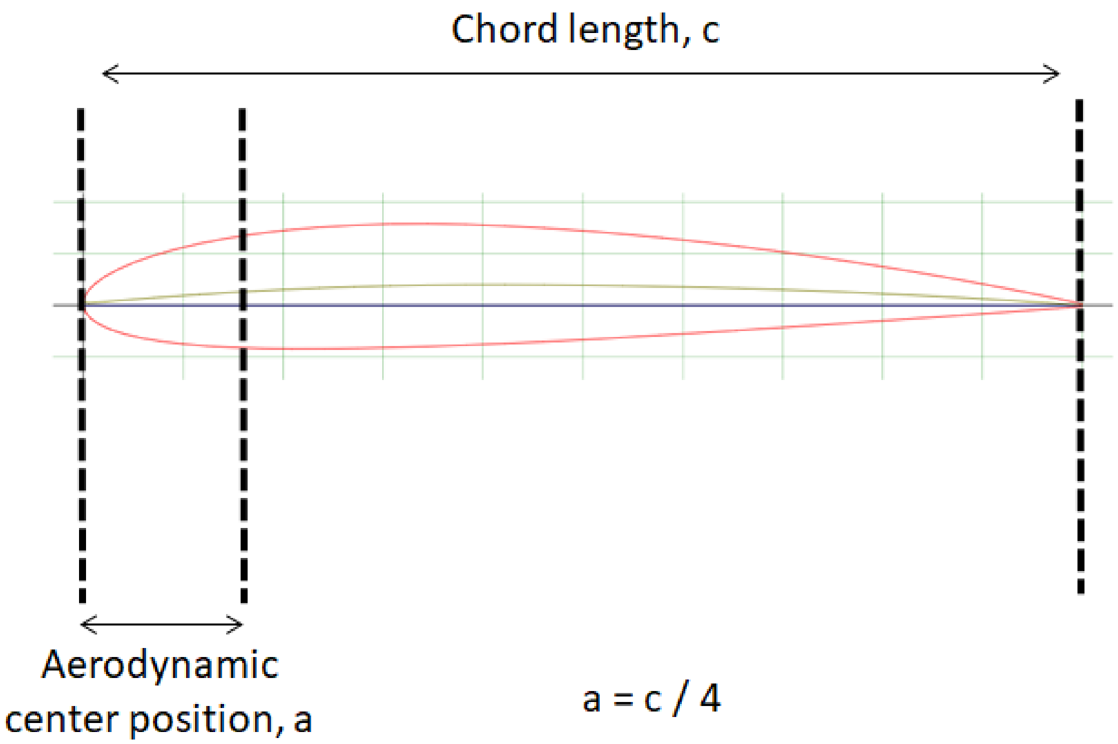


Figure 32. Aerodynamic center of the airfoil

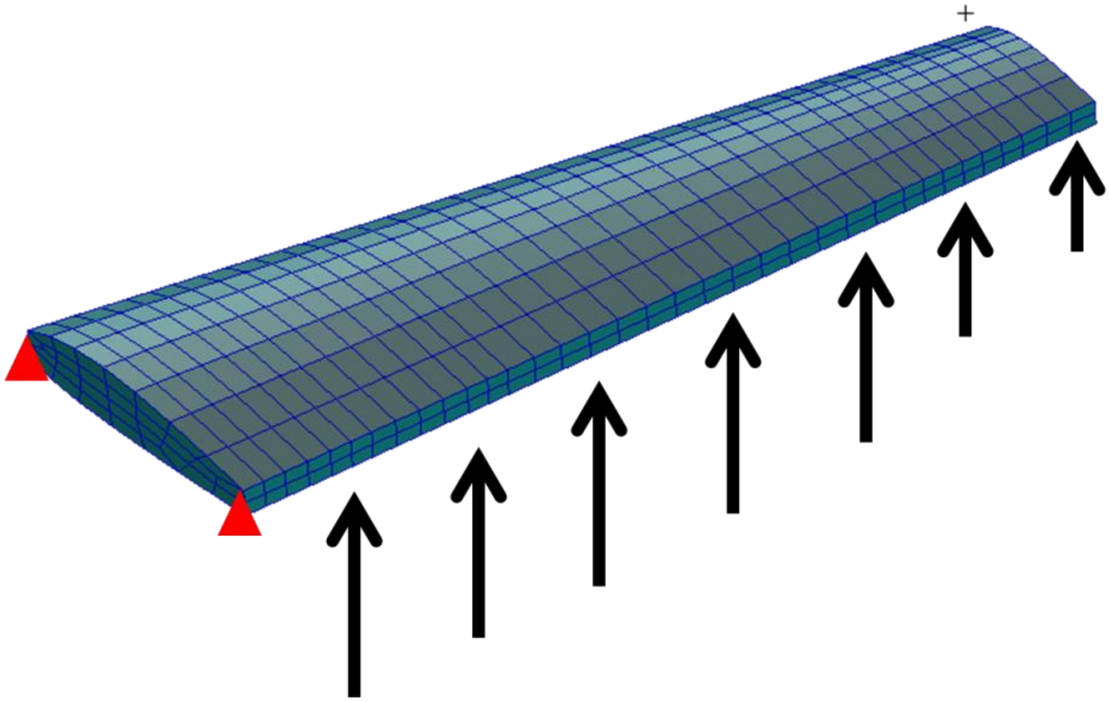


Figure 33. Load case application of FE model

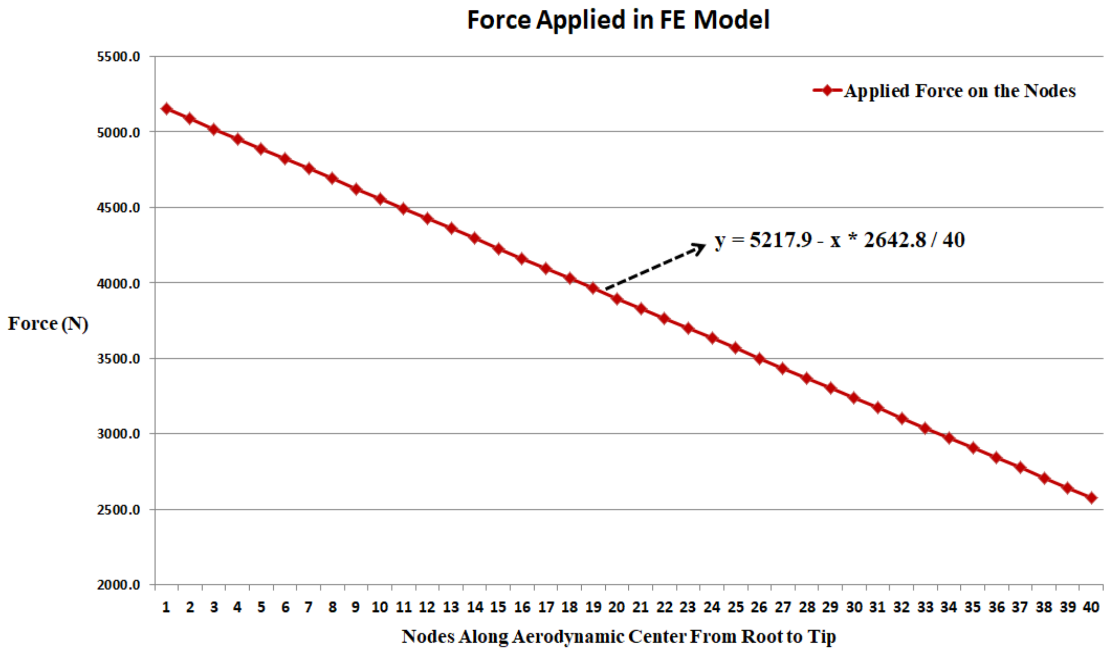


Figure 34. Force distribution along aerodynamic center of the wing box

5.5 General Modeling Principles

5.5.1 Location of Nodes

Nodes of the finite element model are generated at the middle surfaces of the components. Element offsets are defined according to element normal as shown in Table 5.

Table 5. FE model element offset values

Wing Box Component	**Element offset**
Upper Panel	
Lower Panel	
Front Spar	- thickness/2
Rear Spar	
Ribs	

5.5.2 Types of Elements

All components have relatively low thickness values compared to other dimensions. In this case, shell modeling of the components is accepted. 2D modeling of the components is adoptable assumption. Element types and property of the elements are shown in Table 6.

Table 6. FE model element and property description

<table>
<tr><th>Wing Box Component</th><th>Element Type</th><th>Element Property</th></tr>
<tr><td>Upper Panel</td><td rowspan="5">CQUAD/CTRIA</td><td rowspan="4">PSHELL</td></tr>
<tr><td>Lower Panel</td></tr>
<tr><td>Front Spar</td></tr>
<tr><td>Rear Spar</td></tr>
<tr><td>Ribs</td><td>PCOMP</td></tr>
</table>

5.5.3 Element Normal

Element normal is used in thickness and stiffness calculation in finite element analysis. So, it is important and it should be aimed to same sense on the same face of the component. It should be consistent in all components. Each component' element normal are shown respectively from Figure 35 to Figure 38.

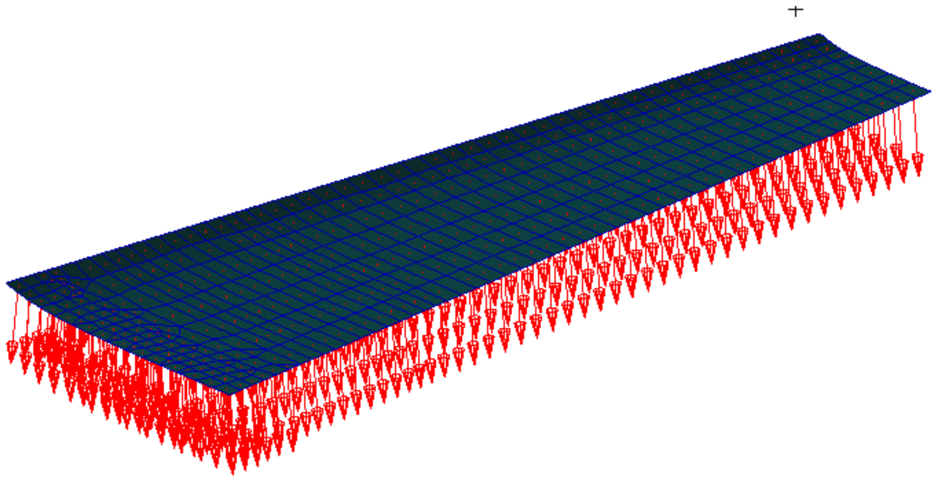


Figure 35. Lower panel element normal representation

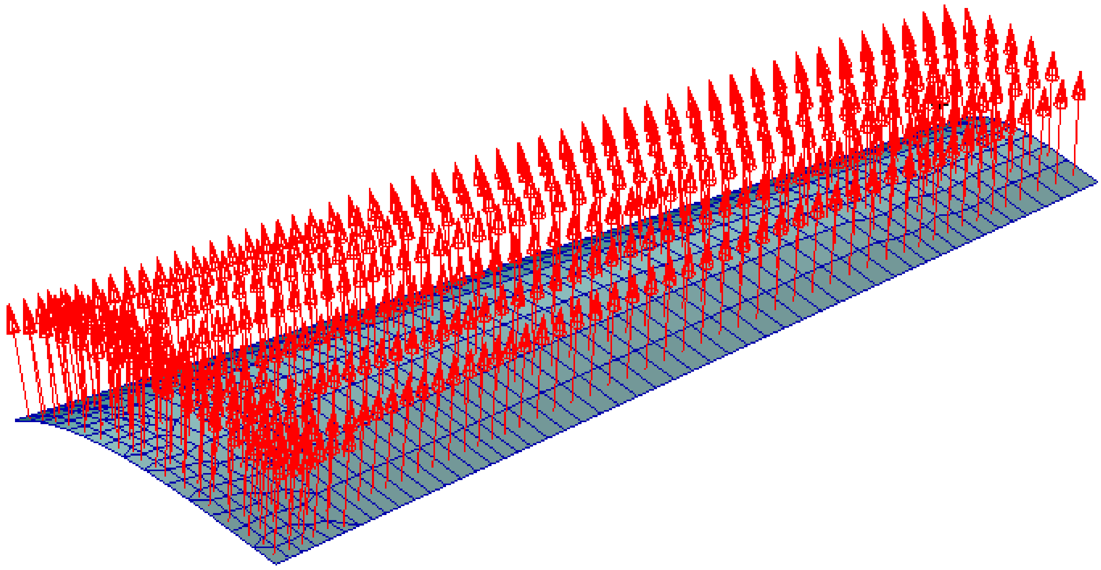


Figure 36. Upper panel element normal representation

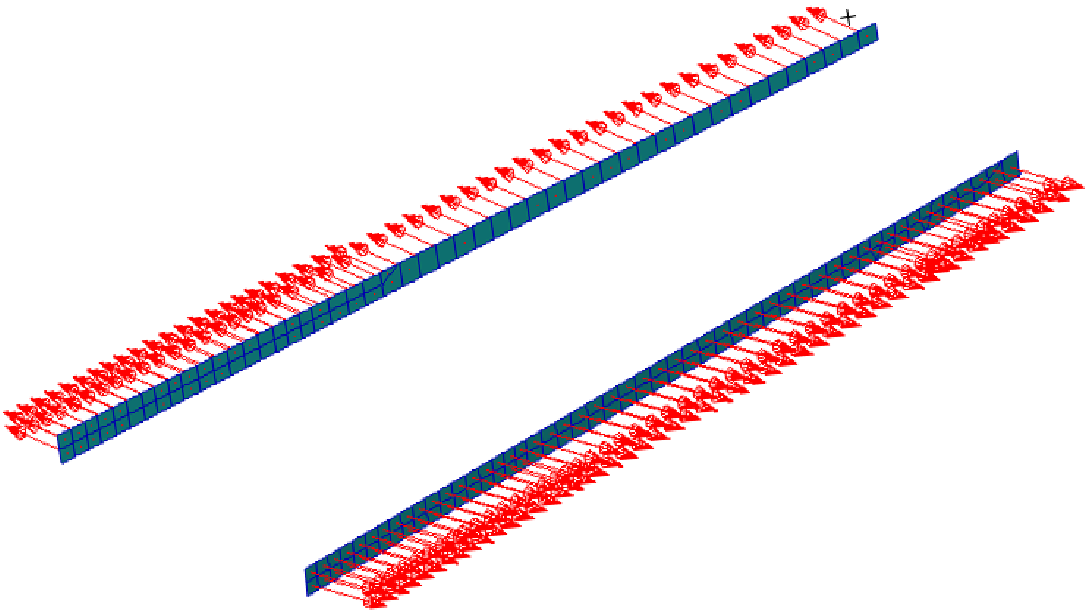


Figure 37. Front and rear spar element normal representation

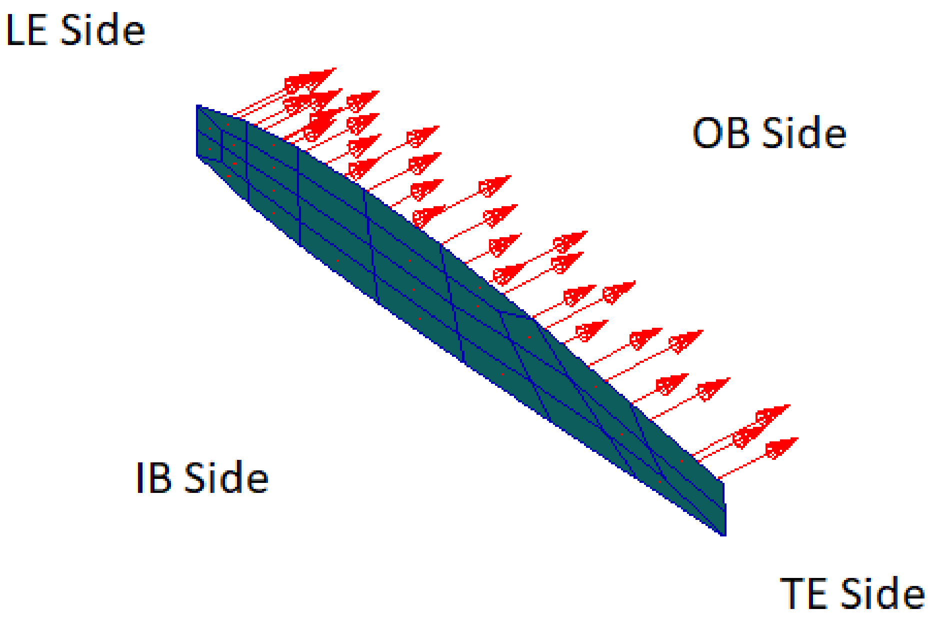


Figure 38. Rib element normal representation

5.6 Material Properties

Two types of materials exist in the aircraft wing box design in the present work. One of them is Al 7475 – T7351 aluminum and the other one is AS4-8552 Hexcel plain weave fabric prepreg 193 gsm composite. The material properties of both are given in Table 7 and Table 8 [4], [21]. Only composite component in the aircraft wing box assembly is rib. Its initial (before optimization) ply stacking configuration is given in Table 9.

Table 7. Al 7475 – T7351 aluminum material properties

Material Name	**Young Modulus (MPa)**	**Mass Density (kg/mm^3)**	**Poisson Coefficient**
Al 7475 – T7351 aluminum	71600	2.7×10^{-6}	0.33

Table 8. AS4-8552 Hexcel plain weave fabric prepreg 193 gsm lamina properties [21]

E_{1t}	64534 MPa
E_{2t}	65569 MPa
E_{1c}	59639 MPa
E_{2c}	59777 MPa
v_{12}	0.046
G_{12}	4964 MPa
cpt (composite ply thickness)	0.198 mm

Table 9. Rib component ply stacking sequence of the initial design

Rib Components Ply Stacking Configuration	45°	0°	45°	0°	45°	0°	45°	0°	45°	0°	45°

5.7 Finite Element Analysis Results

After preparing finite element model of the aircraft wing box design with described BCs, solution is obtained for linear static case using MSC Nastran software. The critical rib component to be optimized is decided according to these results.

Translational deflection plot of the wing box design is provided in Figure 39 in millimeter. It gives translational resultant deflection plot with respect to global coordinate system (Coordinate 0). Wireframe elements represent undeformed body, while colored body shows deformed under applied load and boundary conditions.

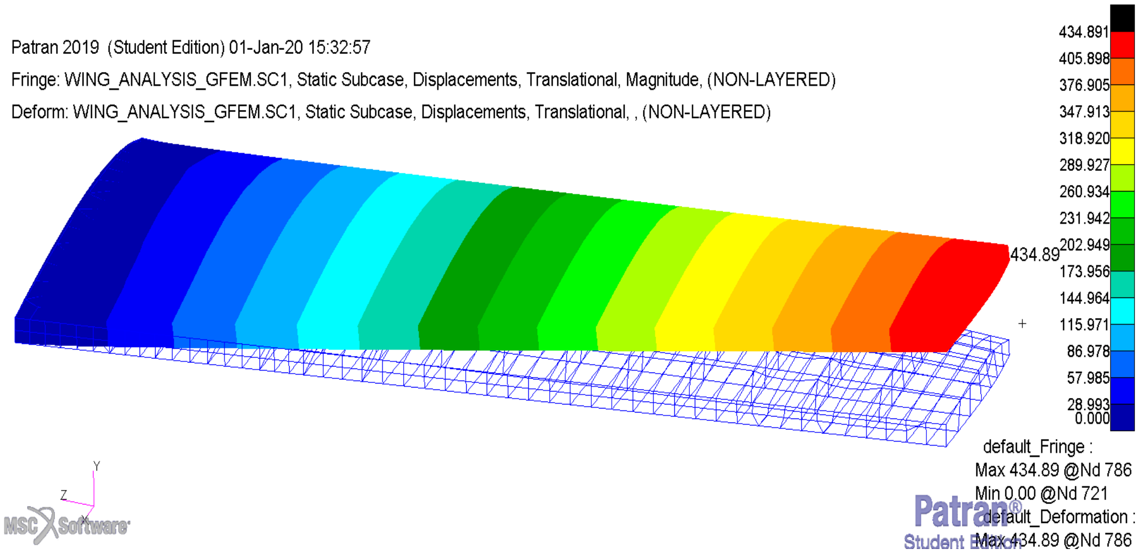


Figure 39. Translational deflection plot of the wing box

After obtaining meaningful finite element analysis deflection results, rib component applied to optimization process is decided. Actually all components should be applied to weight optimization process not only ribs. However due to time limitations, it is aimed to focus on designs consist of composite materials in present work. Moreover, due to time limitations of this study, the process is performed for only the selected critical rib.

Maximum and minimum principle stress distribution of the rib components in magnitude except the closure ribs are given in Figure 40 and Figure 41. Inboard and outboard closure ribs are kept out of the optimization since they are required for the maintenance of aerodynamic consistency of the wing box. From Figure 40 and Figure 41, it can easily be seen that maximum conditions occurs in the first rib at the inboard side. Its maximum and minimum principle stress distributions are given in Figure 42 and Figure 43. It faces with the maximum stress condition among the other ribs.

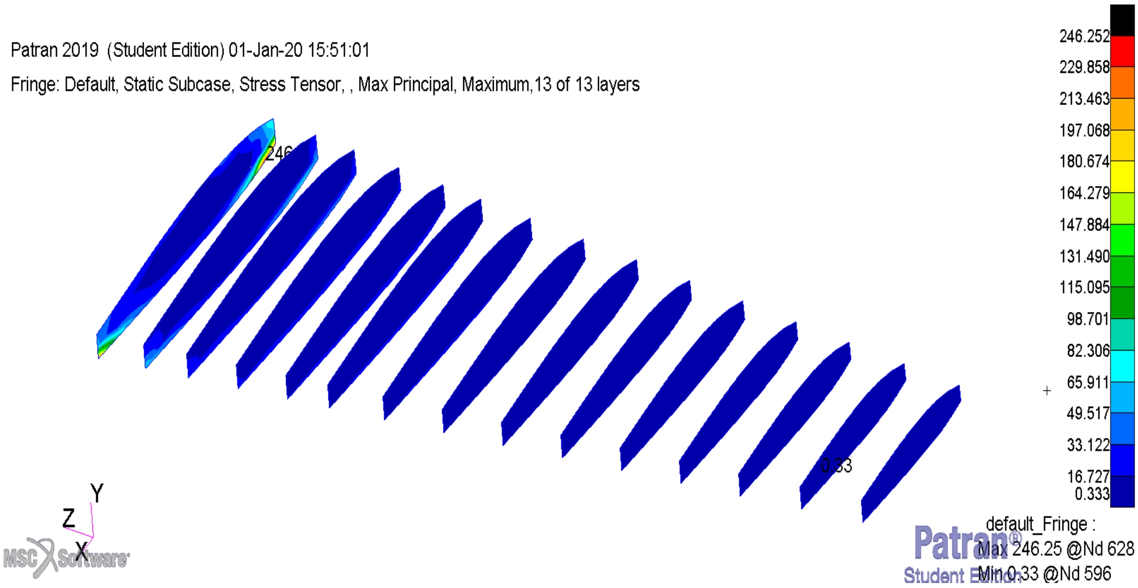


Figure 40. Max. principle stress mapping of the ribs in magnitude

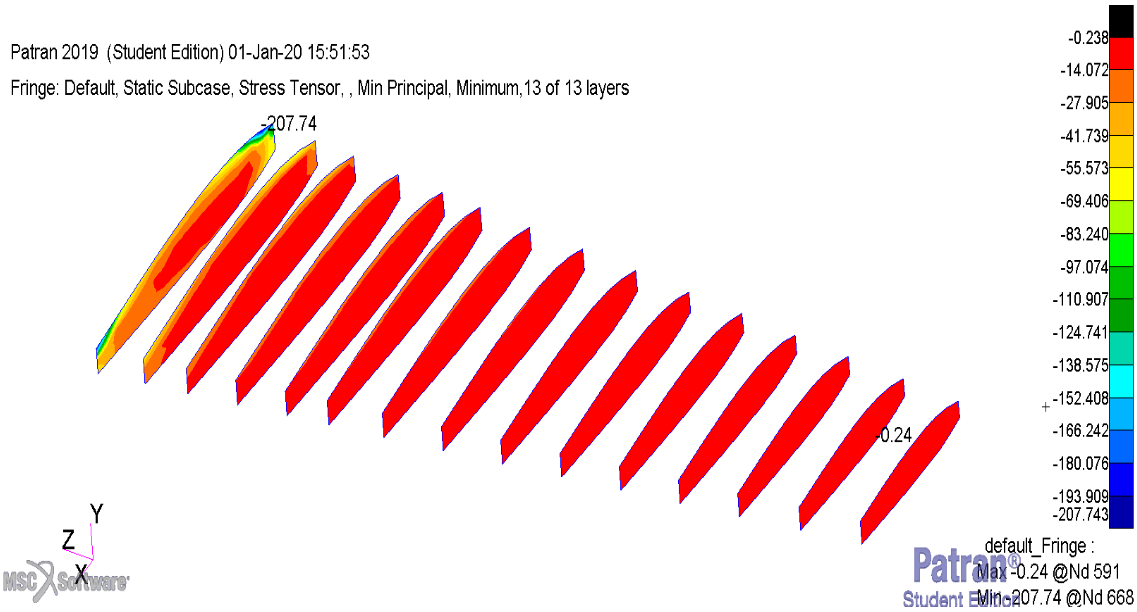


Figure 41. Min. principle stress mapping of the ribs in magnitude

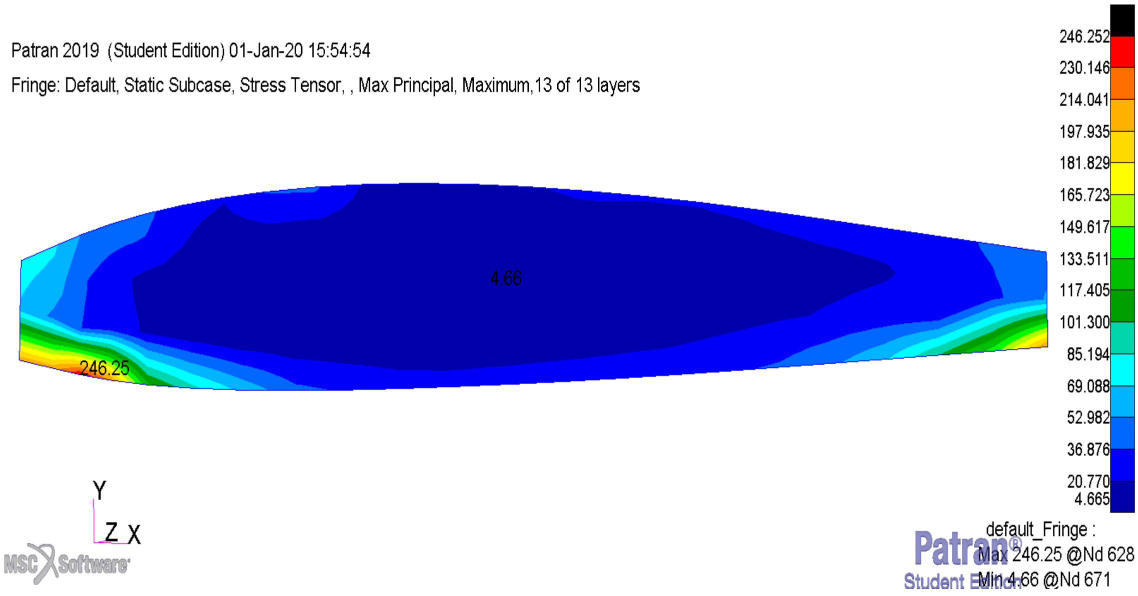


Figure 42. Max principle stress distribution of the first rib in magnitude

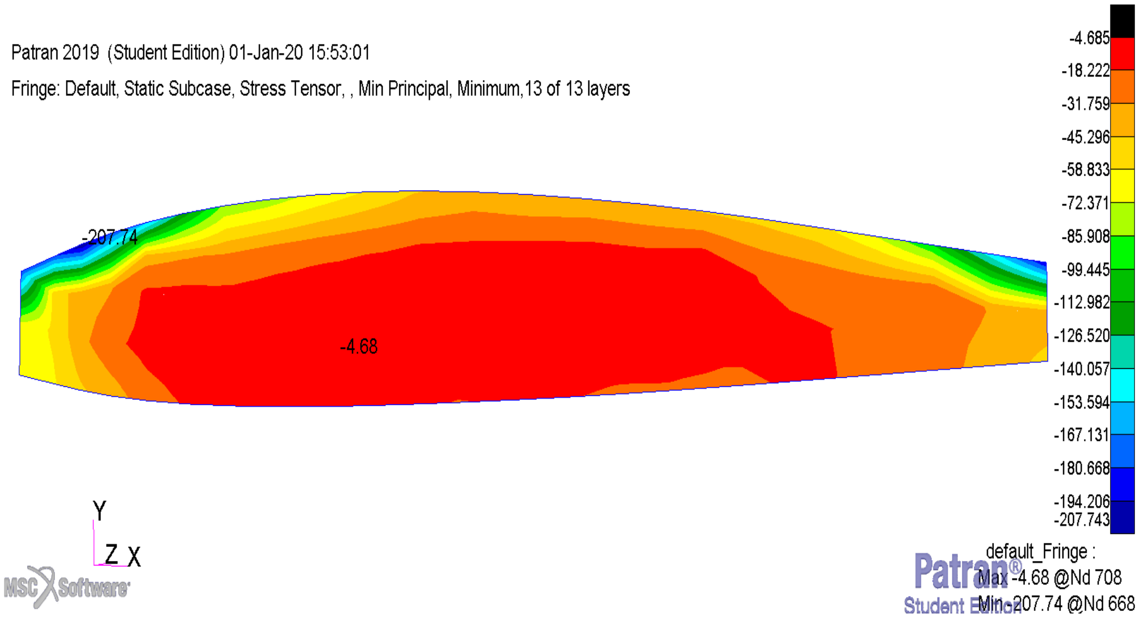


Figure 43. Min principle stress distribution of the first rib in magnitude

Failure index calculation of selected rib design under applied load is performed using Tsai-Hill failure criterion which is one of the phenomenological material failure theories for especially orthotropic materials. Orthotropic material failure strengths of the composite material used in present work are taken from Marlett [21]. The distribution is given in Figure 44. Failure index converges to 1 when design is getting critical by definition and any failure index value greater than or equal to 1.0 is considered failure.

Also maximum failure index of the design is close to 0 when the design is oversafe. From Figure 44, it is seen that overall maximum failure index of the initial design is so small and the initial design is oversafe.

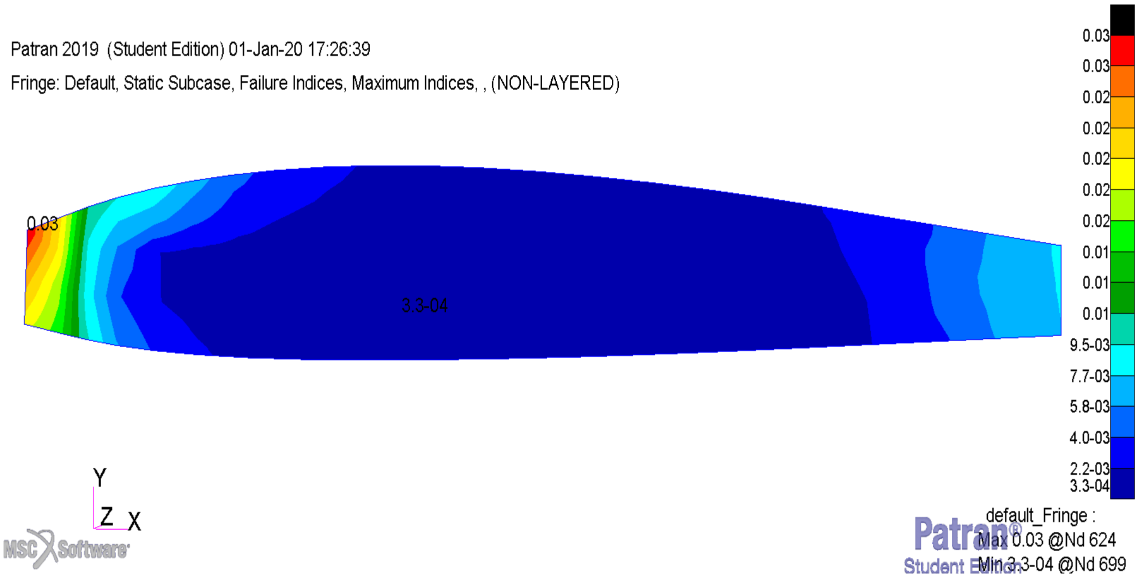


Figure 44. Failure index distribution of the selected rib

5.8 Conclusion

Finite element model of aircraft wing box design is created and analysis is performed in this chapter. The critical rib is selected according to stress distribution on the rib components. Maximum and minimum principle stresses are analyzed for all ribs. First rib at the inboard side except inboard closure rib faces with the maximum stress condition among the other ribs. According to stress analysis of the ribs, first rib at the inboard side except inboard closure rib is selected to be optimized, since optimization process under high level of stress condition is more critical.

6 THICKNESS OPTIMIZATION PROCESS

6.1 Introduction

The workflow of the entire study is presented in Figure 45. The topic explained in this chapter is marked with red rectangle. Firstly, finite element model needs to be changed and prepared for the optimization process in a few aspects. Constant thickness and ply sequence regions should be defined in the finite element model. This preparation is explained firstly. After that, thickness optimization process is applied to the critical rib determined in the previous analysis. Results and outcomes of the optimization process are explained. Lastly, manufacturability consideration of the optimized rib design is criticized.

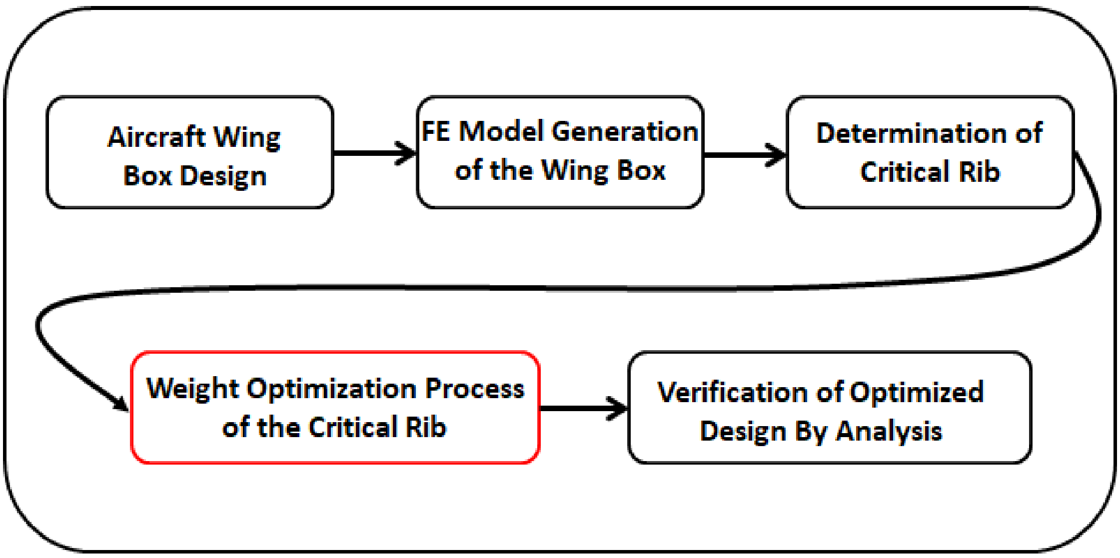


Figure 45. Flow chart of the work

6.2 Model Preparation

The optimization is performed for the critical rib determined in the previous chapter. Hypersizer software is be used in this study as the optimization tool. It takes inputs of finite element model in bdf file type and analysis results in op2 file type. Grid point forces and element shell forces of the solved finite element model are obtained from those inputs. After that, it applies its own optimization method as described in section 3.3 previously.

According to Hypersizer thickness optimization with FEA method, user should divide the component to be optimized into some sections. Those sections represent constant thickness and uniform ply sequence regions. Within these regions, thickness and ply sequence of the composite is optimized together. Also Hypersizer software designs transition regions from one constant thickness region to another one. It takes into consideration manufacturability of the optimized component and minimizes the drop-off in the design. After all those steps, it gives consistent and easy to manufacture optimized components as output.

The constant thickness and stacking sequence regions are decided according to stress distribution on the selected rib component. It was decided to prepare two different division options in the present work. Elements with large stress are tried to collect in a single region for both options. Similarly, elements with lower levels of stress are united in a single region. Although, affectivity of the optimization process increases with the increasing number of constant thickness regions, computational time of the optimization tool and complexity of the manufacturing process of the optimized design increases. It is still possible to obtain high level of weight reduction ratio with low number of constant thickness and ply sequence regions by collecting elements that face with the similar level of shell force or stress together. This is the easier and wisely way of the optimization of components in the Hypersizer software. So, option 1 is divided into four different regions and option 2 is divided into five regions according to stress that they carry. Their regions are demonstrated in Figure 46 and Figure 47.

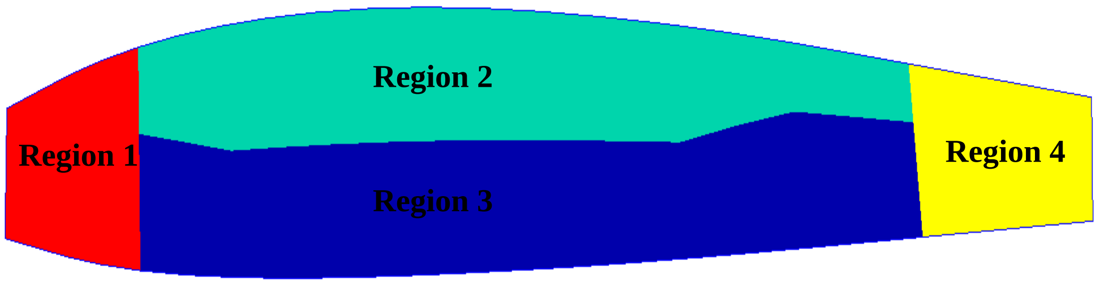


Figure 46. Representation of constant thickness regions of option 1

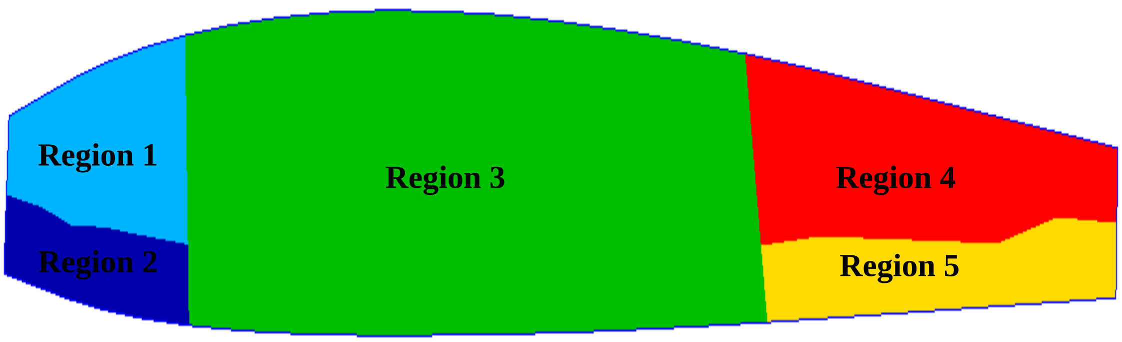


Figure 47. Representation of constant thickness regions of option 2

6.3 Optimization Results

After preparation of the model to the optimization, two different options (option 1 and 2) are implemented in the Hypersizer software. Material and property definitions, initial thicknesses and element offset of the finite element model are obtained from input files. Thus, there is no need to reenter those parameters into the Hypersizer software one more time. After completing input on the tool, the optimization results are obtained.

Initial design of the selected rib component thickness was constant and 2.18 mm with the layers configured as given previously in Table 9. Its total mass was 1.15 kg. Total mass for option 1 after optimization is 0.71 kg. Thicknesses and stacking sequences of all regions for option 1 after optimization are summarized in Table 10. Total mass for option 2 after optimization is 0.83 kg. Similarly, Thicknesses and stacking sequences of all regions for option 2 after optimization are summarized in Table 11. Total mass of all design options and mass reduction percentages of option 1 and 2 are given in Table 12.

Table 10. Optimization results of all regions for option

Region #	# of Ply	Stacking Sequence	Total Thickness (mm)
1	10	[45 / -45 / 45 / -45 / 0]s	1.98
2	6	[45 / -45 / 0]s	1.18
3	6	[45 / -45 / 0]s	1.18
4	9	[45 / -45 / 45 / -45 / 0_]s	1.78

Table 11 Optimization results of all regions for option 2

Region #	# of Ply	Stacking Sequence	Total Thickness (mm)
1	8	[45 / 0 / -45 / 0]s	1.58
2	7	[45 / -45 / 0 / 0_]s	1.38
3	10	[45 / 0 / 0 / -45 / 0]s	1.98
4	6	[45 / -45 / 0]s	1.19
5	5	[45 / -45 / 0_]s	0.99

Table 12 Total mass of all design options

Inıtial Design Mass (kg)	Option 1 Mass (kg)	Option 1 Mass Reduction Percentage	Option 1 Mass (kg)	Option 2 Mass Reduction Percentage
1.15	0.71	38.2	0.83	27.8

Two division options of the optimization process lead two different results as can be seen from Table 12. While option 1 induces up to 39 % weight reduction with respect to initial design, option 2 induces 28 % weight reduction. Results present in Table 12 means that dividing initial design 4 different constant thickness regions is more beneficial for this optimization process with respect to dividing initial design 5 regions. Elements having similar level of stress are united in same regions in option 1 and option 1 is more beneficial than the option 2 for the optimization process.

Table 13 represents 0° and 45° lamina percentages of both optimized designs option 1 and option 2. As can be seen from Table 13, 45° lamina is dominant type in both optimized design solutions. As mentioned previously, rib components in the aircraft wing designs are mainly responsible to withstand against shear load type. So obtaining dominantly 45° stacking sequence results for the optimized designs is expected and mentioned previously.

Table 13 0° and 45° lamina percentages of option 1 and option 2

	Option 1		Option 2	
Region	**0° percentage**	**45° percentage**	**0° percentage**	**45° percentage**
1	20	80	50	50
2	34	66	43	57
3	34	66	60	40
4	12	88	34	66
5	-	-	20	80

6.4 Manufacturability Consideration of Optimized Components

Manufacturability of the composite design can be seen as a biggest restriction in the design stage. There are several different types of composite manufacturing methods and all of them have their own design restrictions. On the other hand, some modern composite manufacturing methods, like automated fibre placement or 3D printing, have low level of restriction in the manufacturing stage of the design components [22]. Through this study, it is assumed that this optimized composite aircraft composite wing rib design is manufactured using conventional hand layup technique.

With the increasing today's technology level, all plies could be cut with laser tools according to design regions and stacking as patched. Then stacking all those patches together desired thickness can be formed.

As shown in Figure 48, plies can be cut according to design and cured together to obtain desired thickness and stacking distributions. In this case, the most important point is transition regions and deciding ply-drops. Hypersizer defines all transition regions and ply drop-offs considering manufacturability rules such as avoiding high ratio thickness change consecutively and so many ramp regions in the design. So that, we could have producible design solutions after optimization stage in Hypersizer.

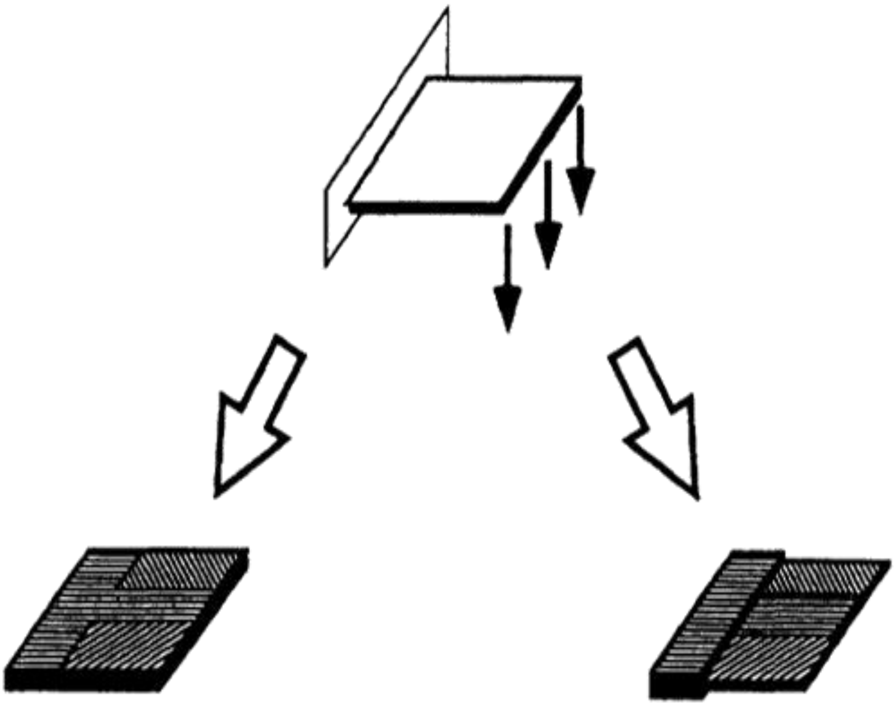


Figure 48. Constant and non-constant thickness distribution of same design [22]

6.5 Conclusion

According to optimization results presented in this chapter, it is seen that weight reduction up to 39 % is obtained by applying optimization method in present work. Moreover, present results show that division constant thickness and stacking sequence regions of the initial design before optimization process highly affects results. Dividing so many regions not always means better optimization results. Initial designs should be divided regions according to load path distributions obtained finite element analysis results. Elements having similar level of stress should be collected in to same regions. This method increases the affectivity of the optimization process. By dividing initial design differently, it is possible to obtain better optimization results. Option 1 leads more effective optimization results in present work and it is selected as result of the study. Although, Hypersizer gives optimization result according to manufacturability rules such as avoiding high ratio thickness change consecutively and too many ramp regions in the design, it is considered in this chapter one more time.

7 FINITE ELEMENT ANALYSIS OF THE OPTIMIZED DESIGN

7.1 Introduction

The workflow of the entire study is presented in Figure 49. The topic explained in this chapter is marked with red rectangle. Optimized design obtained through previous chapters is validated by using finite element method in this chapter. Failure index analysis of the selected optimized design (option 1) is performed firstly. Then, effects on the wing box deflection of this optimization process in present work are investigated. Effects of the optimization process on the wing box are mentioned in detail.

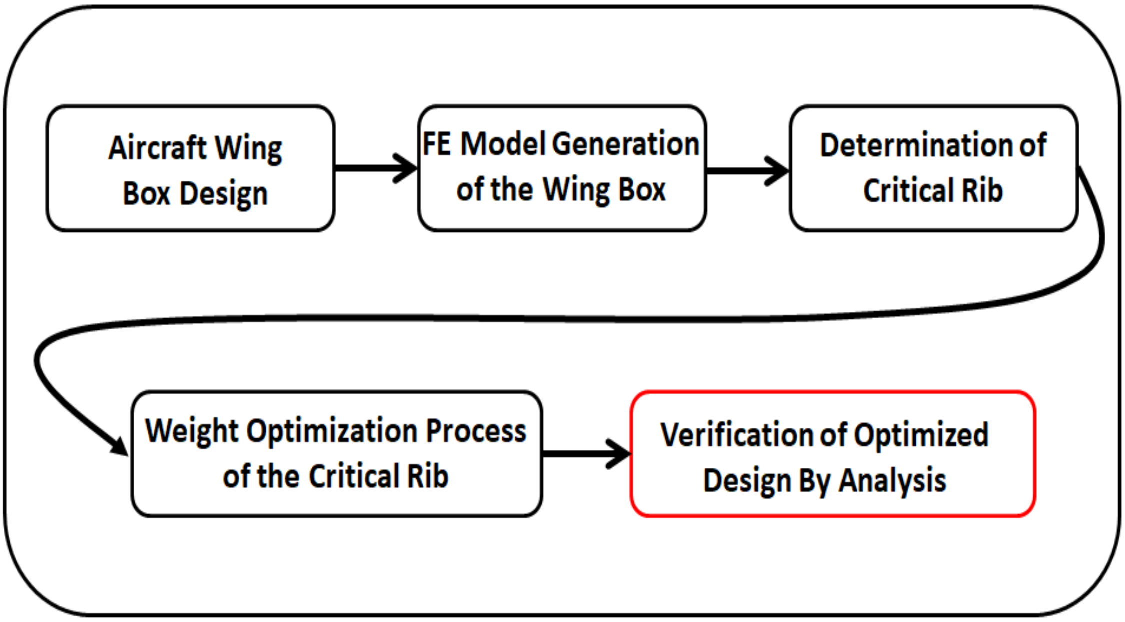


Figure 49. Flow chart of the work

7.2 Analysis of Optimized Design

Finite element model is modified according to results obtained after the optimization process. Thicknesses and stacking sequences of the initial design are changed and finite element model of the optimized design is obtained. Same boundary conditions, load case definitions and material properties are used in the finite element model of the optimized design. Figure 50 represents the failure index distribution of the selected rib. Also, maximum failure index comparison between initial and optimized designs is given in Table 14. Table 14 shows that maximum failure index of the initial design is increased approximately ten times after optimization process. This means structural criticality of the design is increased while total mass of the design is decreasing. Increase in failure index in this amount shows the affectivity of the optimization process. But it is still lower than 1 and in safe margin according to failure condition. Although, maximum failure index of the optimized design can be seen as far away from failure condition (equal to 1), decreasing one more layer in this condition can cause a catastrophic failure scenario.

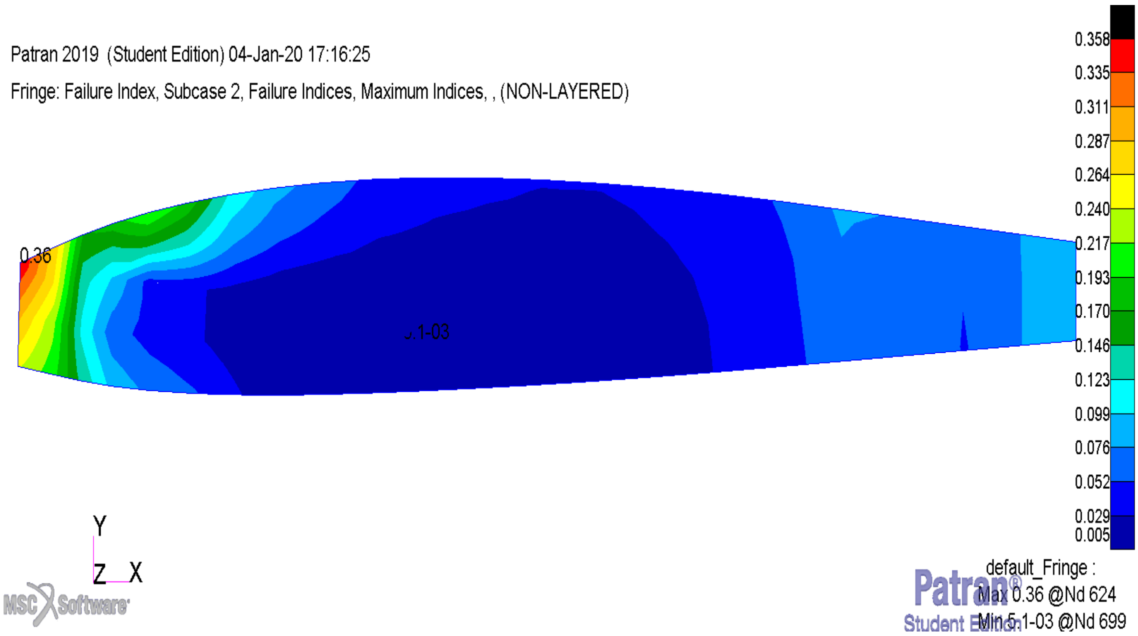


Figure 50. Failure index distribution of optimized design

Table 14 Maximum failure index of initial and optimized designs

	Initial Design	**Optimized Design**
Total Mass (kg)	1.15	0.71
Maximum Failure Index	0.03	0.36

Effect of this optimization process on the overall wing box translational deflection should be investigated. Figure 51 shows the overall wing box translational deflection results after optimization process. Also, Table 15 gives the comparison of wing box deflection before and after optimization. According to Table 15, it is seen that optimization process in present work does not have a huge impact on overall wing box deflection. Translational wing box deflection is changed approximately 1 percent. Results of the optimized design mean optimization procedure applied in the present work made possible to decrease weight of a selected aircraft wing box structural component keeping overall stiffness of the wing box nearly constant as expected.

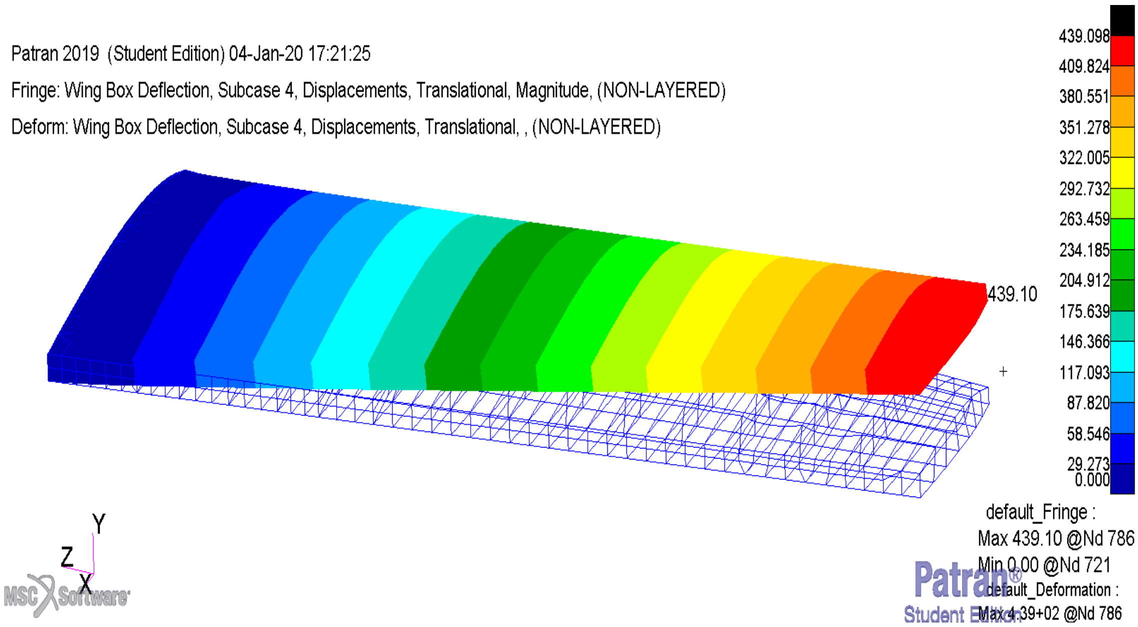


Figure 51. Overall wing box deflection with optimized rib component

Table 15 Overall translational wing box deflection of initial and optimized designs

	Initial Design	Optimized Design
Total Mass (kg)	1.15	0.71
Translational Wing Box Deflection (mm)	434.89	439.10

7.3 Conclusion

Structural analysis of the wing box structure after optimization process is investigated in this chapter. General results and maximum failure index change of the optimized rib component are observed. Possible effects of this optimization process onto wing box structure are obtained. According to results, it is concluded that optimization process applied on the selected rib component in present work increases maximum failure index of the rib and decrease the total weight while keeping nearly constant of overall wing box stiffness. It is shown that optimization process decreased the total weight by keeping constant overall stiffness of the aircraft wing box. It means optimization process applied in the present work is beneficial and advantageous for the structural engineers about weight reduction.

8 GENERAL CONCLUSIONS

8.1 Summary

The method discussed in present work is the weight optimization study on a selected aircraft composite wing rib component. The study started with the design procedure of an aircraft wing to be able to obtain load distribution on the composite wing ribs. At the first stage of the aircraft wing design, airfoil type is selected and outer geometry modeled accordingly. After that, number of spars and ribs, their positions on the wing, their initial thickness values and stacking sequences are decided. Finite element model of the initial aircraft wing box design is prepared. Load case definitions and boundary conditions are decided and applied on the finite element model in MSC Patran software environment. After solving finite element model of the wing box model as linear structural case, one of the structural rib components is selected to optimize and structural weight optimization method is applied on the selected rib component. All through mentioned optimization process, unnecessary, unused sections and oversafe designed regions of the aircraft composite wing rib is specified by using optimization process explained previously. After optimization process, oversafe designed thickness regions are subjected to weight reduction process and thickness are designed thinner locally. Weight optimization and reduction of the selected aircraft composite wing rib is obtained with this procedure. Finite element analysis of the optimized design is performed one more time to observe effects of optimization process on the wing box structure.

In Chapter 2, similar studies in the literature were introduced. Beneficial parts of those researches are explained. In Chapter 3, theoretical background information that helps to understand present study was mentioned. Aircraft wing box design procedure of the present work is explained in Chapter 4. Details about finite element analysis of the initial design of aircraft wing box model are mentioned in Chapter 5. Chapter 6 is dedicated to weight optimization work of the aircraft wing box design. After that, details about finite element analysis of optimized wing box design are explained in Chapter 7.

8.2 Key Findings and Outcomes

The research performed in present study revealed some key findings and outcomes and they can be summarized as follow:

In present work, a method optimizing thickness of a selected aircraft composite wing rib is proposed. Results of the study shows that applied optimization method could be able to find a design solution that is more effective and beneficial than the initial design. It specifies the oversafe designed regions and reduces thickness on prescribed sections. Also this optimization process gives most advantageous stacking sequence of the composite designs and it makes possible more beneficial and light in weight designs in the aviation industry by proposing near-optimum design solutions.

However, conventional design procedure in the aviation industry is conservative. Engineers design a new component according to geometrical needs and after that structural analysts check design whether withstand under applied boundary conditions or not. But local thickness reduction on the component is not generally considered since it is time consuming and laborsome process. Uniform thickness distribution generally considers in the industrial design procedure. This study proposes a general methodology to perform thickness reduction locally on the component and make weight reduction possible in a simplest way. From this perspective, this study can be very useful and helpful in the aviation industry.

Weight is a big problem in the aviation industry. With the increasing total weight of the aircraft, all internal loads are increasing. Lift force needed to maintain aircraft in the air during flight is also increasing with the increasing total mass of the aircraft. With this reason, total mass of the aircraft is very important in aviation industry. Thus, proposed method of present study could be important for the engineers work for aviation industry.

Moreover, structural weight optimization process proposed in present study would be very useful in conceptual design stage in the aviation industry. Conceptual design is very important for many aspects such as initial gross weight estimation used in lift force calculations which highly affects design process of the aircraft. Also, conceptual design is important for total material cost calculations. Total material costs are highly depend on thicknesses and estimating optimized thickness at very early stages of the design process makes possible to estimate total costs of the aircraft correctly at the beginning of the project. This issue is very important for project proposal of the aviation companies. From this perspective, optimization method used in present work would be very useful. Generating concept designs, initial thickness and stacking sequences rapidly can be hard at the beginning of the project for the design engineers. Present method makes possible to define near-optimum design solutions, initial thickness and stacking sequences easier and faster.

8.3 Future Studies

Proposed optimization method in present study works as intended. On the other hand, some improvements can be made to enhance the usefulness and effectiveness of the method. One of these improvements is to automatize the whole procedure. All process from initial design stage to obtaining optimized design can be automatized in another software platform by using any other programing language like Python or C. So, all design stages connected together and it makes easier and faster to design and optimize new structural components. End to end design solution can be obtained for structural engineers with the automatisation of all process from initial design to optimized design by using software tools.

Another improvement could be increasing load case number applied in the finite element analysis stage. In this study, analysis is performed according to single load case which is considered as critical. However, different load cases would be important for the wing box structure and all of them should be used while sizing aircraft wing box structure. More realistic results in the finite element analysis could be obtained with the increasing load case number in the sizing stage.

Moreover, sizing and thickness optimization procedure suggested in present work is applied on the selected single aircraft composite wing rib component and 38 percent of weight reduction is achieved. Proposed optimization process in this study would be applied on almost all structural component of the wing box structure not only ribs. Also it can be applied on to metallic components of the wing box structure. Application to all aircraft component would provide high weight reduction ratios and make more beneficial aircraft designs possible in the aviation industry.

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