ANKARA YILDIRIM BEYAZIT UNIVERSITY

GRADUATE SCHOOL OF NATURAL AND APPLIED SCIENCES

A FURTHER INVESTIGATION FOR PARAMETRIC OPTIMIZATION OF ARCHED AND HONEYCOMB CORED SANDWICH COMPOSITE PANEL FOR AIRCRAFT WING BUCKLING WITH FINITE ELEMENT ANALYSIS

M.Sc. Master of Science Thesis by

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December, 2022

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by

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M.Sc. THESIS EXAMINATION RESULT FORM

We have read the thesis entitled **"A FURTHER INVESTIGATION FOR PARAMETRIC OPTIMIZATION OF ARCHED AND HONEYCOMB CORED SANDWICH COMPOSITE PANEL FOR AIRCRAFT WING BUCKLING WITH FINITE ELEMENT ANALYSIS".** This thesis was completed by **Mustafa BİLGİÇ** under the supervision of **Assoc. Prof. Dr. Metehan ERDOĞAN** and we are to certify that it is adequate as a thesis for the degree of Master of Science.

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December 2022 **Mustafa Bilgiç**

A FURTHER INVESTIGATION FOR PARAMETRIC OPTIMIZATION OF ARCHED AND HONEYCOMB CORED SANDWICH COMPOSITE PANEL FOR AIRCRAFT WING BUCKLING WITH FINITE ELEMENT ANALYSIS

ABSTRACT

This thesis dwells on a further necessity of the parametric optimization work of sandwich composite materials with FEA. The design of sandwich composite materials is not a new area, however, this work will pave the way for the engineers to design and size more effective wing panels, airframe geometries, composite fan blades, and other applications that need reduced weight. In the aerospace industry, sizing is a very important area for lightening the weight of the aircraft. Sizing operation is applied according to the stress and buckling criteria. The thickness of a panel may make the buckling failure mode is the critical failure mode and the buckling is generally the most critical failure mode of thin-walled structures under bending loads such as wings. However, the general applications of sizing are done for metallic materials and simple composite panels. This study explains the optimization studies used in the aviation industry. The sizing applications do not optimize the materials. Sizing operations determine the thickness of the thin-walled structures, however, the different material combinations may have less mass when they are optimized properly. This work strategy optimizes not only the thickness of the materials but also material combinations. Thus, this is a further way for consideration of materials in optimization and sizing. There is a discrimination between this methodology and mostly used sizing operations in the aerospace industry. Generally, sizing operations are done for the selected materials in the aerospace industry, also, the selected materials generally are in stock. However, the methodology used in this research is done for selecting the optimum materials, material combinations and thickness.

SONLU ELEMENLAR ANALİZİ İLE UÇAK KANADI BURKULMASINA YÖNELİK KEMERLİ VE BAL PETEĞİ ÖZLÜ SANDVİÇ KOMPOZİT PANELİN PARAMETRİK OPTİMİZASYONU İÇİN İLERİ BİR ARAŞTIRMA

ÖZ

Bu tez çalışması, sandviç kompozit malzemelerin sonlu elemanlar yöntemiyle yapılan parametrik optimizasyonunun ileri bir gerekliliği üstüne durmaktadır. Sandviç kompozit tasarımı yeni bir çalışma alanı değildir, fakat bu yöntem mühendislere kanat panellerini, havacılık yapısal gövde parçalarını, kompozit fan kanatçıklarını ve diğer düşük ağırlık gerektiren uygulamaları daha etkili bir şekilde tasarlamaları ve boyutlandırmaları için ortam hazırlar. Havacılık ve uzay endüstrisinde boyutlandırma hava araçlarında ağırlığı düşürmek için çok önemli bir uygulamadır. Boyutlandırma uygulaması gerilme ve burkulma kriterlerine göre yapılır. Bir panelin kalınlığı, burkulma kırılma modunu kritik kırılma modu haline getirebilir ve burkulma genellikle kanatlar gibi eğilme yükleri altında ince duvarlı yapıların en kritik kırılma modudur. Bununla birlikte, genel boyutlandırma uygulamaları metalik malzemeler ve basit kompozit paneller için yapılır. Bu çalışma havacılık endüstrisinde kullanılan optimizasyon çalışmalarını açıklamaktadır. Boyutlandırma uygulamaları malzemeleri optimize etmemektedir. Boyutlandırma işlemleri, ince duvarlı yapıların kalınlığını belirler, ancak farklı malzeme kombinasyonları, uygun şekilde optimize edildiklerinde daha az kütleye sahip olabilir. Bu çalışma stratejisi, yalnızca malzemelerin kalınlığını değil, aynı zamanda malzeme kombinasyonlarını da optimize eder. Dolayısıyla bu çalışma malzeme optimizasyonu ve boyutlandırması için farklı ve daha ileri bir yoldur. Havacılık ve uzay endüstrisindeki boyutlandırma metodolojisi ile bu çalışmadaki optimizasyonda kullanılan boyutlandırma işlemleri arasında bir ayrım vardır. Havacılık endüstrisinde genel olarak seçilen malzemeler için boyutlandırma işlemleri yapılır, ayrıca seçilen malzemeler genellikle stokta bulunana malzemeler için yapılır. Ancak bu metodoloji optimum malzemeleri, malzeme kombinasyonlarını ve minimum kalınlığı boyutlandırması için yapılır.

NOMENCLATURE

σ	Normal stress
τ	Shear stress
$1D$	1 Dimensional
$2D$	2 Dimensional
$3D$	3 Dimensional
M	Moment
y	Neutral axis distance
I	Second moment of inertia
q	Shear flow
Q	First moment of inertia
V	Shear force
t	Thickness
e	Honeycomb sheet thickness
h	Cell length
l	Honeycomb edge length
Θ	Honeycomb half inner-angle
ρ^*	Honeycomb density
ρ_s	Honeycomb sheet density
IRF	Inverse Reserve Factor
σ_x	Normal stress on the X axis
τ_{xy}	Shear stress on the XY plane
X^t	Tensile stress limit on the X axis
X^c	Compressive stress limit on the X axis
Y^t	Tensile stress limit on the Y axis
Y^c	Compressive stress limit on the Y axis
S_{xy}	Shear stress limit on the XY plane
E_x	Young's modulus on the X axis
E_y	Young's modulus on the Y axis
E_z	Young's modulus on the Z axis
G_{xy}	Shear modulus on the XY plane
G_{yz}	Shear modulus on the YZ plane

G_{xz}	Shear modulus on the XZ plane
υ_{xy}	Poisson's ratio on the XY plane
υ_{yz}	Poisson's ratio on the YZ plane
υ_{xz}	Poisson's ratio on the XZ plane
E_{sheet}	Young's modulus of honeycomb sheet
G_{sheet}	Shear modulus of honeycomb sheet
υ_{sheet}	Poisson's ratio of honeycomb sheet

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CHAPTER 1

INTRODUCTION

Due to the nature of the aerospace industry, the weight of aerial vehicles has always been one of the biggest problems. Any item on an aerial vehicle that increases the weight is undesirable for aircraft performance. Manufacturers improve their skills in order to make an aerial vehicle as light as possible without sacrificing structural integrity or safety [1].

As much aerial vehicles weigh as less load they can carry, thus, an aerial vehicle must be lighter as much as possible without structural failure. To deal with weight problems, engineers and scientists have been developing complex geometries and advanced materials for centuries [2]. Since the geometry and the material parameters are the 2 ways of dealing with stress as clearly seen in the simple stress formulas below:

Axial Stress Formula [3]:

$$\sigma = \frac{F}{A} \tag{1.1}$$

Where F= Axial Force [N] and A= Axial Area [m^2]

Bending Stress Formula [4]:

$$\sigma = \frac{M * y}{I} \tag{1.2}$$

Where M= Bending Moment [N.m], I= Second Moment of Inertia [m^4] and y= Shear Axis Distance[m]

Shear Stress Formula [5]:

$$\tau = \frac{q}{t} = \frac{V * Q}{I * t} \tag{1.3}$$

Where q= Shear Flow [N/m], t= Thickness Inertia [m], V= Shear Force, Q= First Moment of Inertia [m^3] and I= Second Moment of Inertia [m^4]

As clearly seen in the formulas above, stress depends on the geometrical parameters which are shear axis distance, thickness, the first moment of inertia, the second moment of inertia parameters and the acting force on bodies. To deal with stress under the same load conditions, only geometrical parameters and stronger materials can help engineers.

Since the aim of structural optimization is to lighten the weight of the aerial vehicle in the aerospace industry, geometrical optimization and producing stronger materials ways have been developed for a long time. Stronger and lighter material necessity is the birth of composite materials. Even though composite materials are much more expensive than metallic materials and hard to recycle, they have wide usage areas in the aerospace industry from airframes to jet engine turbine blades due to their material advances over metallic materials. On the other side, geometrical optimizations that are done for weight reduction started with hand calculations with assumptions that depend on the thin-walled structures theory [6].

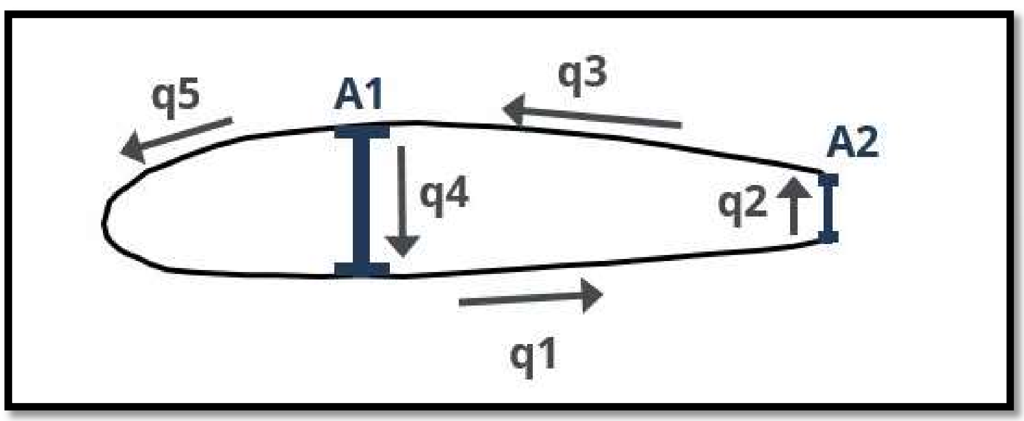

Figure 1.1 Shear Flow in Simple Wing Box Structure [7]

The shear flows in **Figure 1.1** above show the theory depending on some assumptions and finding shear stresses on thin-walled structures. In this theory, thin walls are assumed as they carry the shear load and shear flow (shear force per unit length) is constant along the path. This assumption causes that the shear stress is constant along the path where the wall thickness is constant. Thus, the first moment of inertia that is used in shear stress calculation is obtained with the spar flange area and its distance from the shear axis so this assumption does not make engineers have an optimum design. However, this assumption is used even today in preliminary design stages because of its ease of use and good results.

With the accumulation of the test results and powerful computers and developed software programs, today engineers can optimize airframes better and reduce the weight more than assumption including ways. 2 types of finite element modeling are mostly used in the aerospace industry. One of them is GFEM and the other one is DFEM. GFEM (Global Finite Element Modeling) is done for load extraction on components and the extracted load results are used in hand calculations. GFEM is done with big meshes because the aim is load extraction and fast results. DFEM (Detailed Finite Element Modeling) is done for critical points and detailed investigation in that points such as stress concentrations. DFEM is done with smaller meshes when compared with GFEM and it is generally used in fatigue and damage tolerance analyses. Buckling sizing operations in the aerospace industry are done with the help of GFEM analyses with self-developed and/or commercial software programs such as HYPERSIZER.

1.1 Buckling

Buckling is a structural failure mode such as yielding. It is a sudden change in the shape of a structure. Buckling has no permanent failure effect on structure; when the load is removed, the shape of the structure will return to its undeformed shape. However, buckled structures lose their structural integrity and after buckling they can face plastic permanent failures. Buckling can be seen in large, long and thin structures more easily since the buckling depends on effective length and moment of inertia. Thus, it is the most critical failure mode in airframes because airframes consist of thin and large panels. Also, buckling is a mathematical phenomenon opposite to the other structural failure criteria. The long history of buckling theory for structures began with the studies of Euler [8] in 1744. This study depends on the 1-D compressed beams. The panel buckling introduction was formulated by Von Karman in 1910 [9], that introduction considers linear elastic plates. The post-buckling theory was introduced by Koiter in the 1940s [10-12], Koiter's works consider imperfection and critical stress reduction. The mathematical theories of stability and bifurcation were dwelled in the works of Poincar´e, Lyapunov, and Schmidt [13-15].

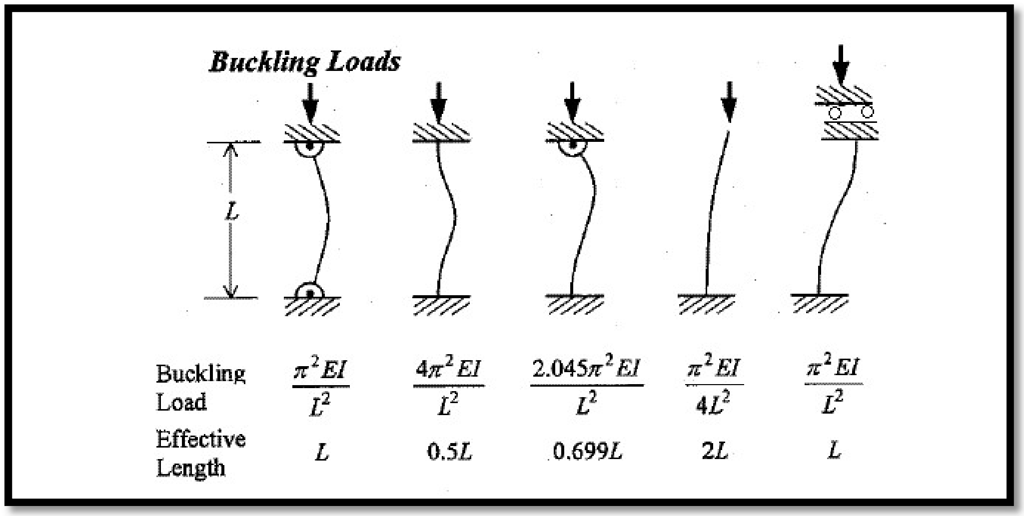

Figure 1.2 Buckling Load and Effective Length of 1D Simple Beams [16]

Panel bucking differs from beam buckling due to that length and width change the results in the panel buckling. The critical load formulas for panel buckling are much more complex when they are compared with beam buckling. However, the finite element analysis with an eigenvalue buckling solution eases analyzing the buckling situation without exact formulas. The shell panels are widely used for light aircraft vehicles, these panels are straight finite planes and they have a geometric configuration where one dimension that is thickness, which is much thinner than the plane dimensions [17]. A shell panel might be assumed as thin when its transverse shear deformations are negligible compared to its bending deflections that include out-of-plane deformation. [18] A compression member exhibits structural instability when it is unable to withstand rising loads and instead shows a reduction in load-carrying capability. The issues with instability can be split into two groups: those involving the bifurcation of equilibrium and those in which the instability arises when the stress on the structural element reaches its maximum or ultimate value without bifurcation. A perfect structural component would initially deform in one way when subjected to increasing compression axial load, but at a certain load known as the critical buckling load, its deformation abruptly shifts to another pattern. The structural components in the latter group, however, experience deformation in a single-mode from the beginning of loading until they reach the ultimate [19].

1.2 Airframe Sizing

Most airframes deal with thin panels or shells which may have buckling characteristics under critical loads. This buckling characteristic may end up in catastrophic failures on the buckled panel or load transferred panels or joints. To prevent these failures, the shell panels are sized to a critical thickness with a margin of safety under consideration of hundreds of load cases and load extraction on panels [20].

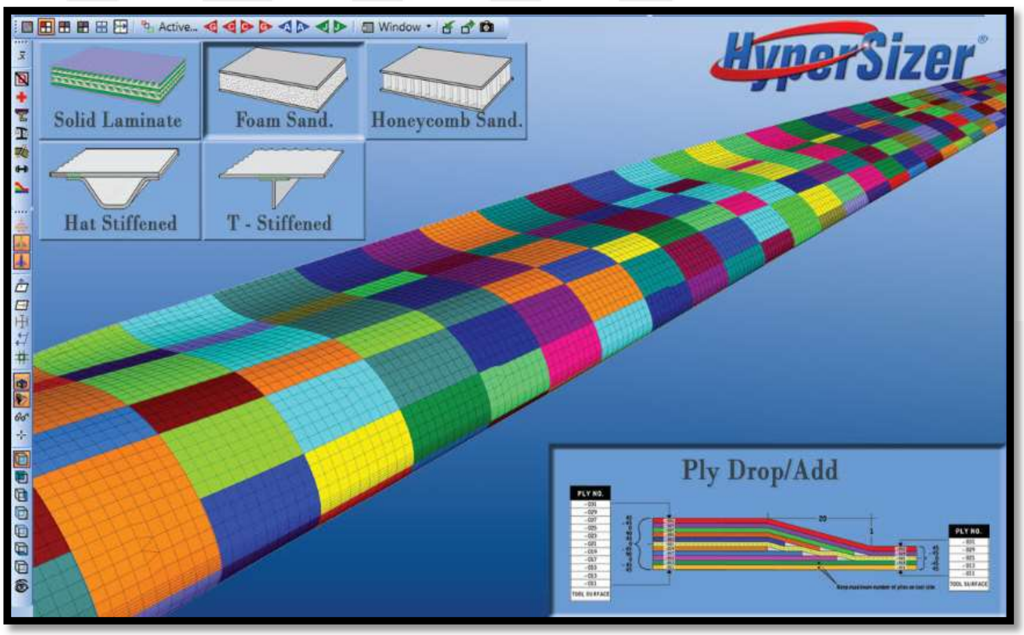

Figure 1.3 Airframe Sizing with HyperSizer [21]

As seen in **Figure 1.3**, the airframe is divided into components with relatively bigger-sized meshes for stress consideration. Those components are sized according to their critical loads. The sizing operation is completed by extracting the most critical load on the sized panel. The bigger meshes are used for only load extraction. Since the FEA considers the nodes for displacement and force reaction, small meshes are not necessary for the load extraction due to the huge number of load conditions [22]. After the sizing operation, the airframe thickness will be determined according to both critical thickness and the manufacturing capabilities. This operation will size the thickness for a given material, however, it does not select the combination of materials. Thus, it is not a fully developed optimization process and there will be a necessity for iteration steps for stress evaluation. This software uses exact solution formulas for buckling sizing. For exact solutions, there should be test results because exact solution formulas have some parameters that were obtained from test results. This exact solution way is very fast for the simple airframe geometries since there is a great accumulation of the test results that were examined by the aerospace companies and exact formulations have been extracted according to these results with some coefficients. However, this sizing operation differs from optimization software programs. In this thesis, I have completed my optimization with ANSYS Mechanical and its optimization tools. For buckling estimation, the FEA solution integrated eigenvalue buckling estimation was used. This way has some pros and cons in the exact solution process. For the geometries that were not tested and have unknown buckling parameters, this way is very useful to have optimum thicknesses without testing. Also, in this way material optimization is possible, namely, optimization progress can decide the combination of materials.

1.3 Pre-preg Laminated Composite Material Design

Pre-preg laminated composite materials or pre-impregnated laminated composite materials are made of fibers and a partially cured polymer matrix such as epoxy or phenolic resin, or even thermoplastic mixed with liquid rubbers or resins. The pre-preg fabrics are generally made of carbon fiber or glass fiber materials and those fabrics are uni-directional anisotropic materials. The lamination is completed by stacking up the fabrics in the “Z” direction of the shell panel rosettes with different angles according to the load condition.

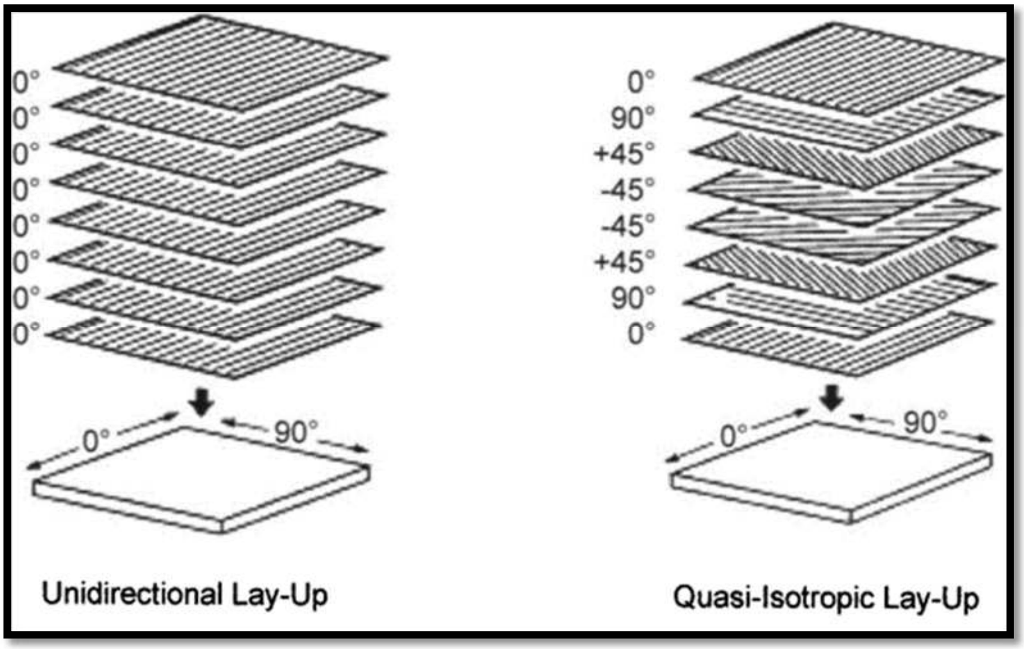

Figure 1.4 Two Types of Pre-preg Stack Up [23]

If the load condition is applied from only one axis. The lamination is completed by stacking up the lamination on one angle. The pre-preg fabrics are strong in only one direction and this direction is the vertical installation direction of carbon fibers. The other directions are weak when they are compared with the vertical installation direction. Because of this, there are some different types of composite pre-pregs.

The design process involves all the decisions certain to produce, operate, maintain, and sell a product. The design begins with the recognition of needs. Satisfying these needs is the problem of the design engineers. Performance criteria are defined and considered at this phase, namely, any solution proposed later will have to satisfy them to be considered an acceptable solution. Problem definition leads to statements about potential solutions and specifications of the different kinds of ingredients that may participate in the solution of the design problem [24]. Also, as seen in the **Figure 1.4**, quasi-isotropic lamination is completed. The below half and above half same angles from the half of the composite and counting to the tips. This symmetrical lamination has advantages on the A-B-D compliance matrix of the composites. This will be explained in the following chapters.

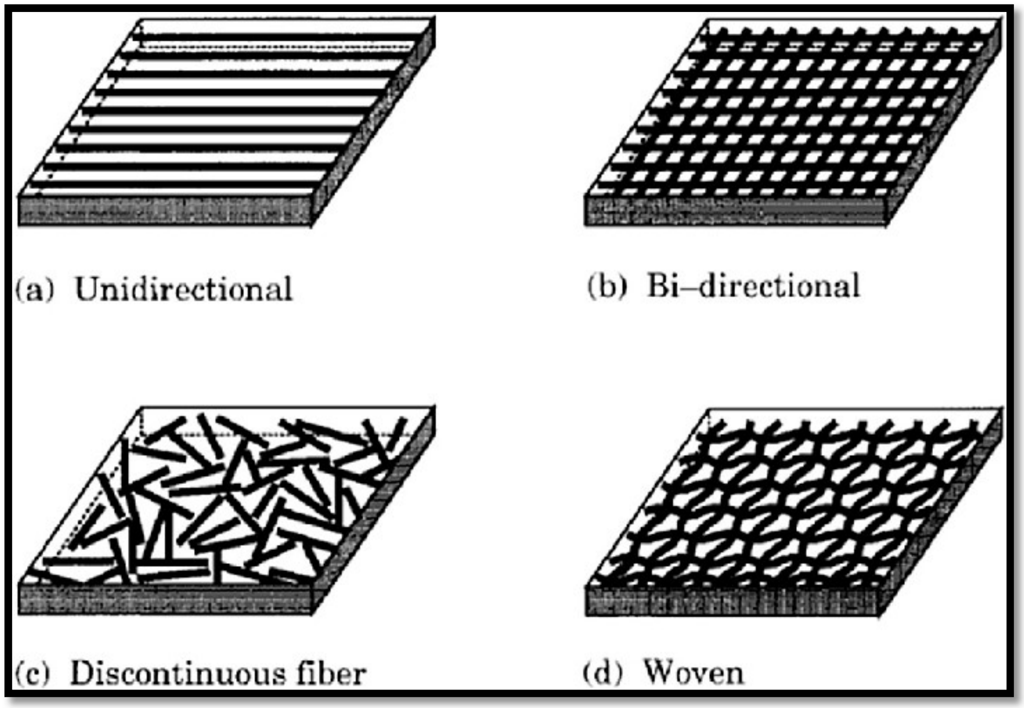

Figure 1.5 Four Different Types of Pre-preg Fabrics [25]

The lamination with different angles for these types of composite materials is completed according to the load conditions. The lamination angles and the fabric densities in different directions must be designed according to the load conditions in order to have the minimum weight.

1.4 Honeycomb Core in Sandwich Composite Materials

Honeycomb core in sandwich composite materials is a very light and strong material. The main idea of the using honeycomb core is to thicken the composite shells and increase the second moment of inertia without becoming the material heavier. Since the honeycomb geometry is the best way when the aim is using the least material for plating a surface. There are 2 common types of honeycomb material for high strength. The first one is aluminum and it is used with aluminum face materials. The second material is a derivative of polyamide. It is poly-aramid which is a high-strength polymer and it is used in bulletproof vests. Honeycomb cores that are made of poly-aramid materials are generally used with laminated composites.

Figure 1.6 Honeycomb Core in a Sandwich Composite Material [26]

Honeycomb cores have orthotropic elasticity when they are considered from the macroscopic side, however, the materials in them have isotropic elasticity. This elasticity changes with some parameters of the honeycomb geometry and material property. A honeycomb geometry can be defined with the parameters below:

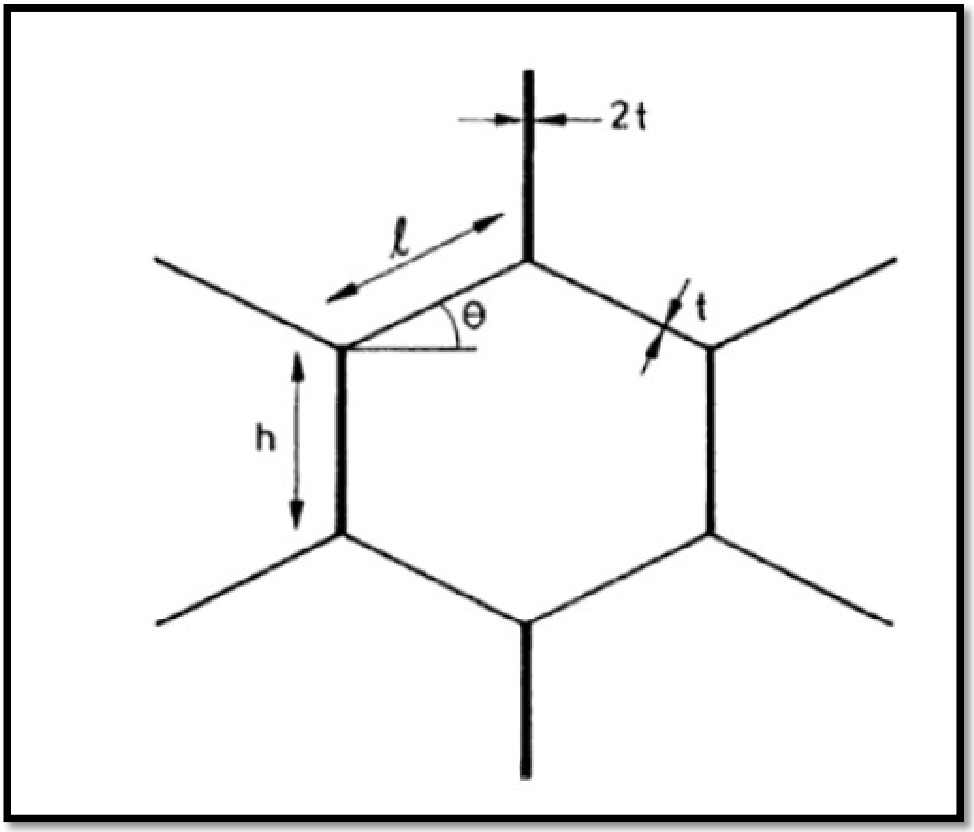

Figure 1.7 Honeycomb Definition Parameters [27]

Honeycomb definition parameters are vertical wall length: “h”, inclined edge length: “l”, half inner angle: “θ”, and sheet thickness: “t”. Commercial honeycombs are produced by combining the sheets. Thus, the vertical walls are 2 times thicker than the inclined walls.

1.4.1 Density of Honeycomb

$$\frac{\rho^*}{\rho_s} = \frac{\frac{t}{l} * (\frac{h}{l} + 1)}{\left(\frac{h}{l} + \sin\theta\right) * \cos\theta} \tag{1.4}$$

As seen in the **Equation 1.4**, the RHS of the equation belongs to a hexagonal-shaped honeycomb having 3 edge connections and 6 edges. Thus, most of the cores in sandwich composites are hexagonal shaped. Then the inclined and vertical wall length is equal. Also, for the value of t = $(\frac{3\sqrt{3}}{8}h)$, the honeycomb is fulfilled. Then, the honeycomb density is equal to the sheet density for the hexagonal honeycomb. There are other useful formulas for Young's moduli, Poisson's ratios, and shear moduli in the longitudinal, transverse, and lateral directions. However, the production of honeycomb cores is not perfect. Thus, the calculated values may not match the test results as shown in **Section 3.1.2** [28].

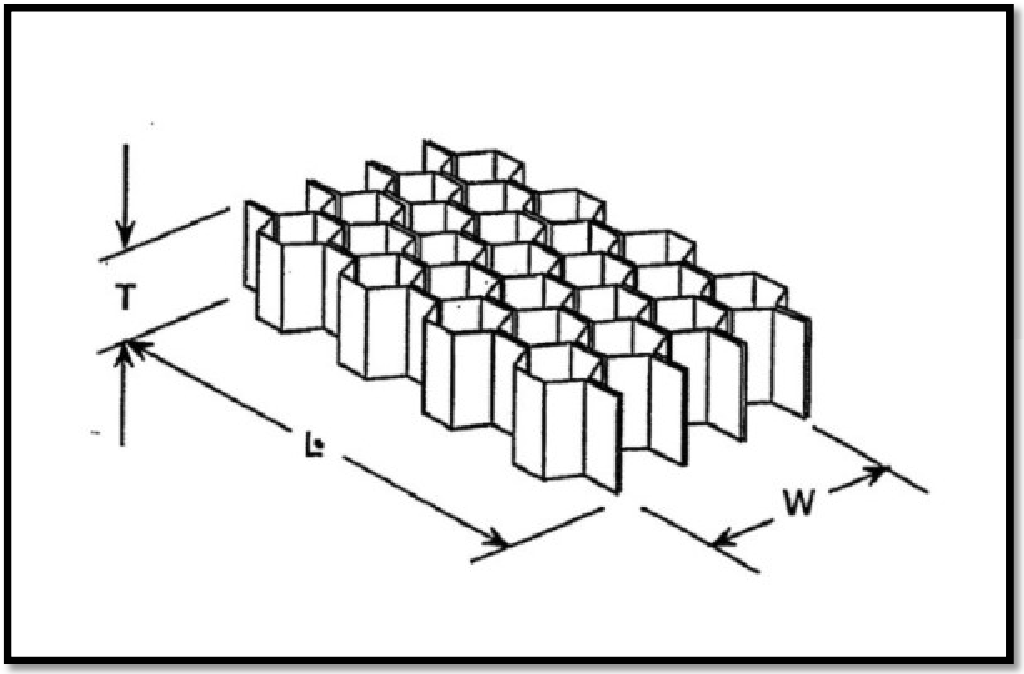

Figure 1.8 Honeycomb L, W & T Directions [29]

Honeycombs are defined with the L, W, and T directions according to their stiffness properties. Also, in finite element modeling, the software considers the L direction as the X direction, the W direction as the Y direction, and the T direction as the Z direction.

1.5 Sandwich Composite Materials and Their Usage in Airframes

The idea that inspires the sandwich composite design is to increase the moment of inertia without increasing the weight as much as the moment of inertia, as dwelled on by D. Gay with a simple numerical application [30]. The sandwich composites are designed in order to handle a special situation. This combination creates a material that is specialized to do a certain job, especially, to become stronger, lighter, or more resistant. They can also improve the strength and stiffness of airframes in the aerospace industry. Sandwich composite material is a developed composite material made of 3 layers with a stacking way. They are generally a combination of a low-density core and 2 thin face panels. They are bonded to each side with an adhesive that has very low stiffness. Sandwich composite panels are generally used in large applications where a combination of high structural rigidity and low weight is required such as aircraft wings.

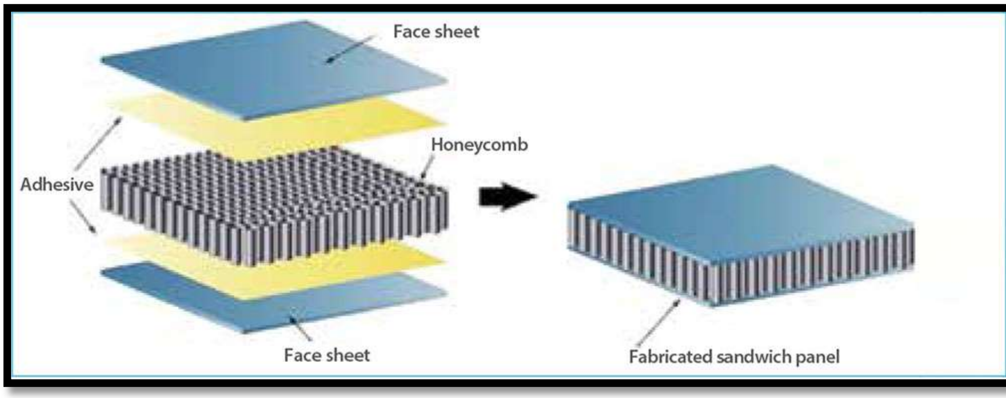

Figure 1.9 Sandwich Composite Material [31]

Face and core materials can be made of different materials. Face materials generally are made of carbon fiber laminated composite and aluminum materials.

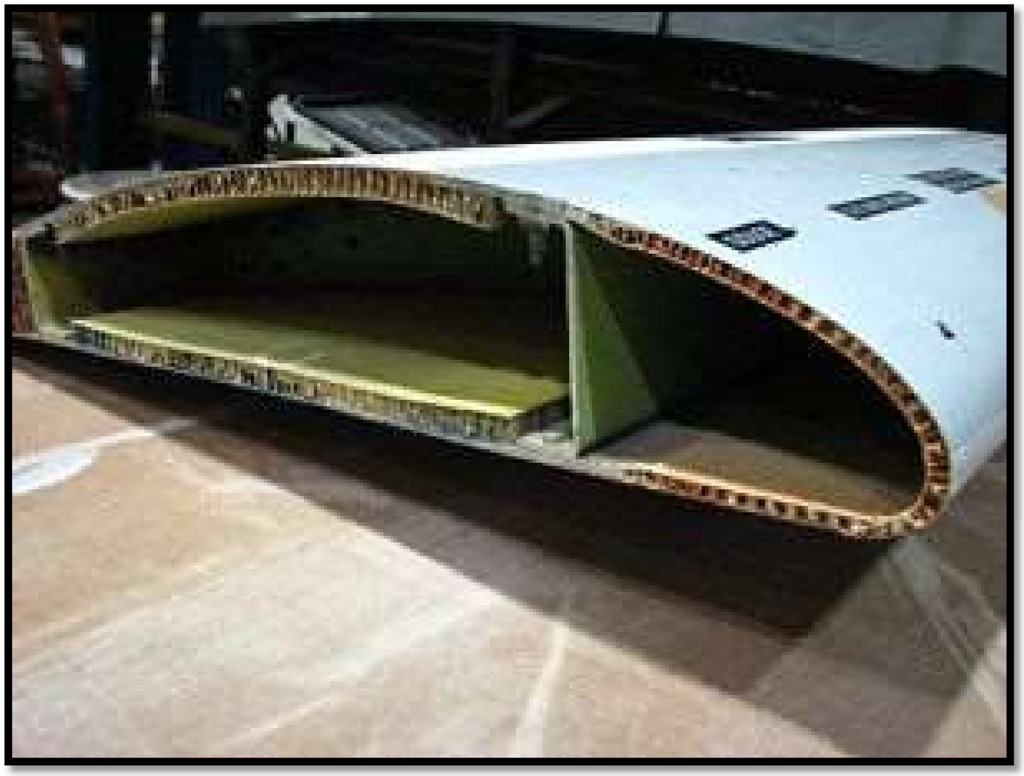

Figure 1.10 Application of Sandwich Composite Material on Aircraft Wing [32]

Sandwich composites are designed for lightweight and high stiffness on aircraft wings. The main purpose is to prevent buckling on the airframe. When they handle the buckling, the simple moment of inertia consideration is the most effective thing in that phenomenon just like an “I” beam. That geometry is used in nearly all of the engineering applications that handle moments.

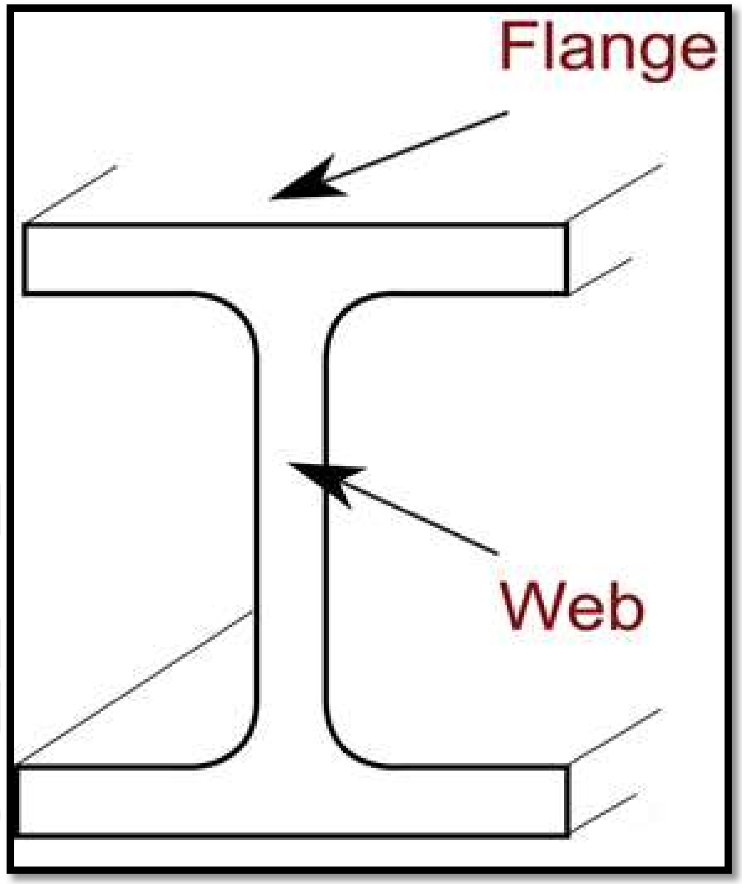

Figure 1.11 I-Beam Schematic [33]

As seen in the **Figure 1.11** above, most of the area of the I-beam is the flange section. When the web section length increases, the moment of inertia created by the flange section increases as the square of that elongation since the area becomes distant from the neutral axis. For the sandwich composite materials, the idea is the same. When the face materials become distant from each other, the moment of inertia increases. And this inertia improvement increases the stiffness that resists the bending moment. Also, sandwich composite material is very useful for the shear loads. The honeycomb core distributes the shear load properly from the source panels to the target panels. This distribution of loads prevents the local stress concentration in continuous panels. The monocoque and semi-monocoque structures are good examples of the idea of the honeycomb effect for stiffening and preventing buckling without any honeycomb core. In addition, the project time plans tend to decrease along the existence period of industries. The time plans and the durations in phases of the automotive industry and the aerospace industry differ from each other because the methodology ripenesses are different. The new application duration changes the time plans because of this reason. New applications may not be completed by the ripe methodologies. For example, the James Webb Space Telescope, which is the newest telescope application of NASA, have different types of time plan sag since it is one of the greatest engineering application and it could not be done with classical methods and methodologies. They are found during the project. Since the sandwich composites are being used for a long time, the sizing and optimization methodology must be faster. This is the main idea and inspiration of this thesis.

The linear elastic behavior of the sandwich composite materials is developed in the literature [34-36]. The airframe buckling sizing formulas depend on these formulas. However, the core of the sandwich composite material needs to be designed with special attention since allowable stresses are low for these sections [37].

1.6 Optimization & Optimization Matrix

The "optimization" word means the action of making the best or the most effective use of a situation or resource. The optimization methodology for structural integrity in the aerospace industry aims to provide the best design in a series of combinations and parameters in terms of lightweight and high strength. Different parameters and combinations can be described in a matrix for optimization. All of the different parameters or combination inputs create a new dimension and expand this matrix. To have the best combination of all parameters, there are different methods to select this combination. These methods can be classified into 2 categories. One of them is the global search and the other one is the local search. Popularity in the process of optimization has taken a giant leap with the improvement of the digital computer in the early 1950s. Recently, optimization processes advanced rapidly and considerable signs of progress have been achieved. Simultaneously, HPC clusters became faster, more versatile, and more efficient. Consequently, it is now possible to solve complex optimization problems which were thought demanding only a few years ago [38].

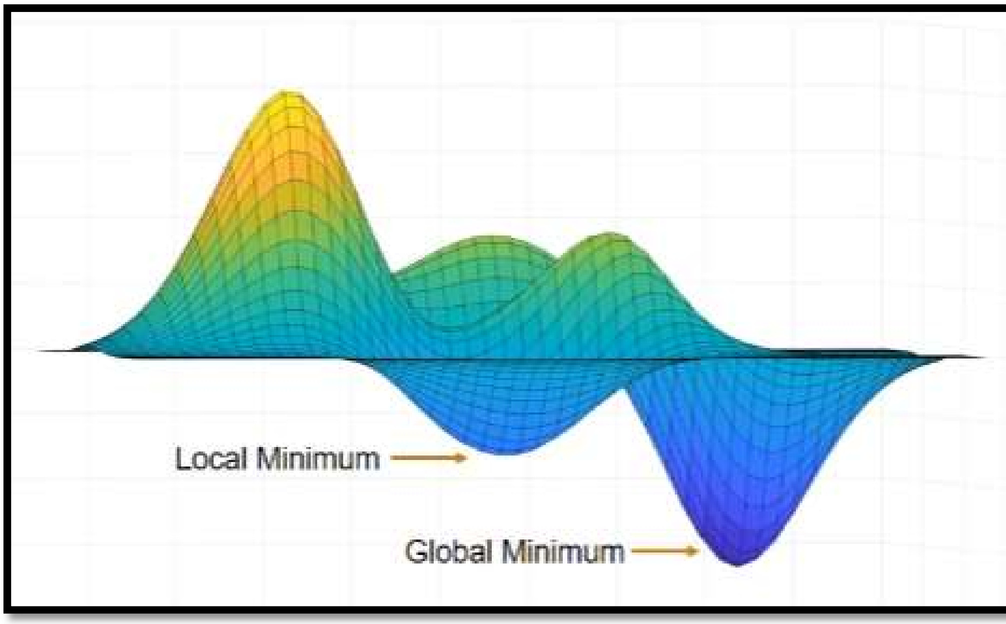

Figure 1.12 Global and Local Minimum Points on a Global Optimization Matrix [39]

In a global optimization matrix that is consisted of different parameters such as in this thesis work, there are a huge number of local minimum and local maximum points. To consider all of the cases, the global search must be completed since considering a local point may give a wrong result in the application. To deal with this problem, the optimization process considers global search with cleaning out the saddle point generating arguments in the matrix. The saddle points occur due to material property change for this case.

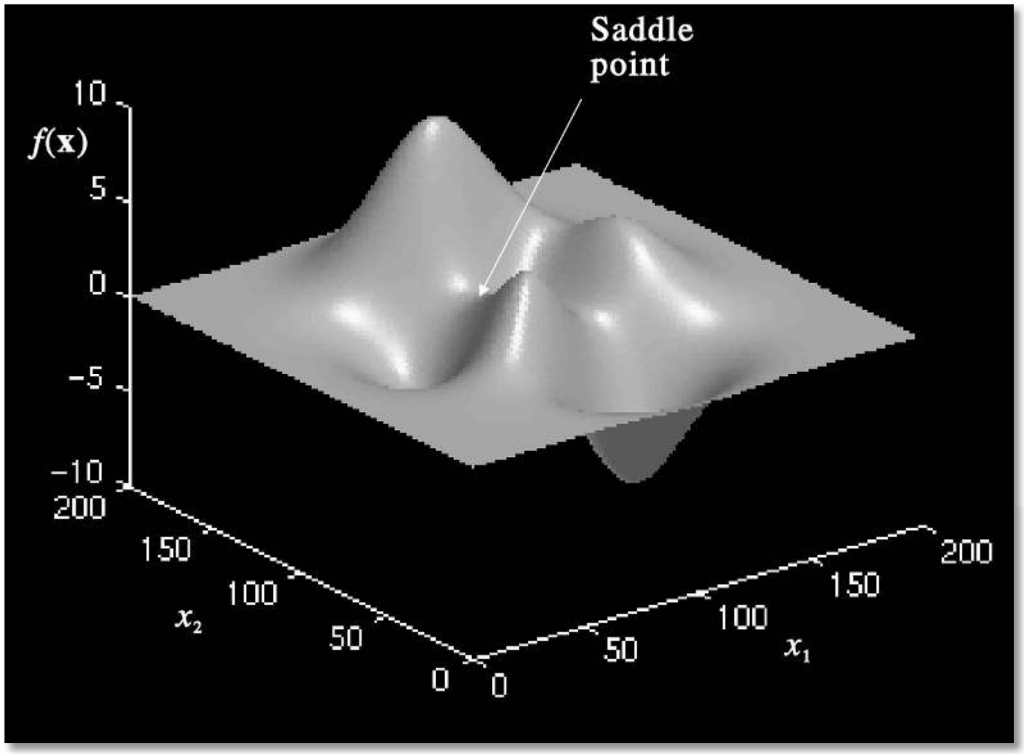

Figure 1.13 Saddle Point on a Global Optimization Matrix [40]

To eliminate the saddle points and generate an effective algorithm, the material properties are arranged according to their stiffness/weight ratios. To complete this arrangement and elimination constant geometrical properties with different material combinations were analyzed. Then, the global matrix is reduced into a local matrix and local research was completed by changing geometrical properties.

1.6.1 Parametric Optimization

Parametric optimization is an optimization process that tries to find the best parameter combinations in the optimization matrix. The parameters in the optimization matrix can be dependent or independent. Ravichandran, Masoudi, Fadel & Wiecek dwells on those problems with specific design variables acting as implicit functions of independent input factors are solved via parametric optimization. For the full parameter space of interest, the ideal answers and ideal values for the objective function are given as functions of the input parameters. General non-convex and non-linear problems require approximations because exact solutions to parametric optimization problems are only possible for those that are linear or convex-quadratic [41]. The optimization process in this thesis have high number of dependent variables. These parameter generates an implicit and complex function to solve. In order to handle this, parametric optimization is used.

1.7 Aim of the Thesis

The aim of this thesis is to generate an optimization process for complex design of sandwich composite materials. This complex design consider stress failure and buckling failure and these 2 are the most critical failure problems in thin walled structures. The optimization processes that are completed with optimization tools may converge incorrect optimization parameter values due to high number of dependent variables. Optimization response function gets more complex when the parameter number increase. In order to converge the correct results parametric optimization is used and an optimization process considering buckling and stress failure is generated.

CHAPTER 2

LITERATURE REVIEW

This thesis dwells on the parametric optimization of the sandwich composite materials, the papers and works about the whole of this work do not consider the whole of the optimization process. Namely, the material engineering side of composite materials research is generally about developing and producing better, stronger and cheaper materials. The optimization side of the composite design is about simple laminated composite panel fabric angle optimization. The whole engineering aspects of this work are considered and completed by the aerospace companies, especially, the aircraft body manufacturers such as Airbus use the sandwich composite materials effectively in their wide commercial aircraft bodies. Their workflow and specialized tools are not shared, however, the parametric optimization programs and FEA are very useful to design sandwich composite materials for academic works and the engineers who do not have special tools. Also, the sandwich composite materials are started to be used in different applications such as aircraft engine casing. This work aims to have a new optimization process that considers material optimization that has not been dwelled on so far and easy process without in-house tools. The optimization processes used before are generally consider fixed material for multi-objective optimizations by changing geometrical properties. However, this methodology decides which combination of material and optimized geometry under multi-objectives. For simple panels, optimization might not change the weight results abruptly, but, for an extra wide-body aircraft such as Airbus A-380, the total weight change would be remarkable with this method. The honeycomb and face materials will be optimized with this method to have less mass in total.

2.1 Composite Materials & Application Areas

A composite material is a type of material that is produced by combining two or more materials. This combination is done to give a unique combination of the properties of the included materials [42]. The advantages of the composite materials are that they usually provide the best material characteristics of the constituent materials and some features that neither constituent possesses. These properties of the composite materials can be used to improve the structural integrity, rigidity, corrosion resistance, wear resistance, attractiveness, weight reduction, fatigue life, thermal insulation, thermal conductivity and acoustical insulation properties of the composite combinations [43].

The composite material area has been existing since the ancient ages. Human beings have been using composite materials in different areas since ancient times. For example, one of the earliest composite materials is Mongolian arcs, which are pieces made of corn and wood, as well as cow tendons, which are combined together with glue [44]. On the other hand, the modern composite technology generation may have begun in 1937. A salesman who had been working in Owens Corning Fiberglass Company started to sell fiberglass to interested parties around the United States. In 1930, fiberglass had been invented. It happened almost by accident, when an engineer became curious about the fibrous material that was created during the process of applying lettering to a milk container.

The application areas of composite materials are very large. Some of the main application areas of the composite materials are listed below [45]:

- Electronics
- Buildings
- Road Transportation
- Rail Transportation
- Marine Transportation
- Air & Space Transportation

2.2 Composite Lamination Angles in Wing Structures

The laminated composite materials in airframe geometries do not have much complex angle orientation due to some manufacturing problems. In this thesis, the optimization was completed by using the 0°, ±45° and 90° pre-preg angles.

Laminated composite materials in wing structure are a combination of laminates having different angles such as 0°, ±15°, ±30°, ±45°, ±60°, ±75°, and 90°. The wing configuration is a hard and iterative process. The selection and calculations are typically repeated several times. An outstanding suite of tools and software based on the previous ten years of aerodynamic and numerical methods, with a reduction there, the number of iterations is observed [46].

2.3 Laminate Stiffness Matrix (A-B-D Matrix) of Composite Materials and Symmetric Lamination

The laminate stiffness is the matrix that is used for the calculation of the stress-strain behavior of the composite materials in the classical laminate theory. This matrix considers the extension stiffness, extension-bending coupled stiffness and bending stiffness of the laminate.

$$\begin{bmatrix} N_x \\ N_y \\ N_{xy} \end{bmatrix} = \begin{bmatrix} A_{11} & A_{12} & A_{16} \\ A_{12} & A_{22} & A_{26} \\ A_{16} & A_{26} & A_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{bmatrix} + \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{bmatrix}$$

$$\begin{bmatrix} M_x \\ M_y \\ M_{xy} \end{bmatrix} = \begin{bmatrix} B_{11} & B_{12} & B_{16} \\ B_{12} & B_{22} & B_{26} \\ B_{16} & B_{26} & B_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{bmatrix} + \begin{bmatrix} D_{11} & D_{12} & D_{16} \\ D_{12} & D_{22} & D_{26} \\ D_{16} & D_{26} & D_{66} \end{bmatrix} \begin{bmatrix} \kappa_x \\ \kappa_y \\ \kappa_{xy} \end{bmatrix}$$

Figure 2.1 Laminate Stiffness Matrix

The stiffness table is composed of different stiffness components for different directions. The "A_{ij}" terms refer to the extension stiffness of the laminate, the "B_{ij}" terms refer to the bending-extension coupled stiffness of the laminate and the "D_{ij}" terms refer to the bending stiffness of the laminate [47].

2.4 Sandwich Composites

Sandwich composites are made of a light core and 2 thin face materials that are placed on the 2 sides of the core. Sandwich composites are used in many applications due to their lightweight and high flexural strength properties. The core material reduces the overall density while rigid layers provide strength. Also, there are adhesive layers that combine different materials. In this way, both materials' weak properties are covered and strong properties are revealed [48].

The core forming the sandwich structure resists the shear stresses and creates a rigid structure by providing uninterrupted support to the surfaces. Moreover, this maintains the distance between the outer surfaces. This distance provides high rigidity on the cross-section in terms of torsional rigidity and moment of inertia. The internal adhesive bonds between the surfaces and the core prevent deterioration and strengthen the flexibility property of the sandwich composite. There is no general rule between core thickness and face thickness. This relation depends on the required features. The biggest advantage of the sandwich structure composites is selecting component materials and their volumetric fiber density and their adjustability [49].

Adhesives that combine the sandwich composite structure provide structural integrity for shear and normal stresses with the help of an uninterrupted connection between the core and the face panels [50].

2.5 Nomex Sheet Based Honeycomb

Nomex honeycomb is a form of Nomex sheet. Nomex sheet is made of kevlar instead of cellulosic fibers. In addition to that Nomex honeycomb is widely used in aerospace applications, it is getting much more popular in other areas due to its great mechanical and thermal properties. On the other hand, Nomex honeycomb is much more expensive when compared with different honeycomb materials such as aluminum and thermoplastic honeycomb [51].

2.6 Aluminum Based Honeycomb

Even though aluminum honeycomb is cheaper than the Nomex honeycomb, they have similar stiffness and strength. The marine applications of aluminum honeycomb may have corrosion problems. The surface gathering applications that are done with carbon should be completed with caution. To prevent the delamination, the surface-core bond must be maintained and this can be achieved by using quality adhesives [52].

2.7 Thermoplastic Based Honeycomb

Thermoplastic-based honeycombs generally have low density and low rigidity when compared with Nomex and aluminum-based honeycombs. Their polymer materials can vary and some of the most used materials are ABS, polycarbonate, polypropylene and polyethylene. They are resistant to many chemicals and do not get affected by humidity. The main problem of this type of honeycomb is making the bond between the surface and the core [53].

2.8 Plies

Plies are the sub-parts of a composite structure. They can be sorted as uni-directional, bi-directional, woven fabric and FRP (fiber-reinforced composite). A ply is consisted of a matrix and fibers. They are also known as laminas or layers of composites. The continuous fibers in the laminas may have different orientation directions. In the airframe structures, the plies can be in different forms, those are prepreg form or dry fabric form. The common form is the prepreg form. This form is preimpregnated (laminate have resin before the production) by fibers and a resin matrix in a semi-cured state. Namely, the fibers and the resin are combined, however, they are still pliable in order to provide easy production. The prepreg form is used in large aircraft body structures. The dry fabric form is the form that consists of dry fabric layers that are impregnated (laminate does not include resin) with a resin matrix that is uncured and has a low viscosity value. This is known as the "*wet layup*" process. For a uni-directional ply, the fiber orientations are aligned in a single direction. For woven fabrics, the weaving in the ply bends the fibers and the fibers are nor closely packed. This bend and packaging reduce the static strength and rigidity of the woven ply for compression loading. On the other hand, the stiffness increase for the tensor loading because of the fiber stretching [54].

2.9 Composite Failure Theories & Tsai-Wu Criterion

The composite laminates' structural mechanics are much more complicated. This complexity cannot be considered by only one failure criterion, hence, there are tons of failure criteria for composite materials. Some of them are maximum stress, maximum strain, Hoffman, Tsai-Hill, Tsai-Wu and Puck failure criteria. These criteria are mostly used and older ones. All of them have shortages when compared with the real results. However, Christos Kassapoglou dwells on that Tsai-Wu failure criteria have prediction results with acceptable errors when compared with test results and the formulation is shown like the below formula [55].

$$\frac{\sigma_x^2}{X^t X^c} + \frac{\sigma_y^2}{Y^t Y^c} - \sqrt{\frac{1}{X^t X^c}\frac{1}{Y^t Y^c}}\sigma_x\sigma_y + \left(\frac{1}{X^t} - \frac{1}{X^c}\right)\sigma_x + \left(\frac{1}{Y^t} - \frac{1}{Y^c}\right)\sigma_y + \frac{\tau_{xy}^2}{S^2} = 1 \tag{2.1}$$

For anisotropic materials, there is a generalized failure theory called as "Gol'denblat and Kapnov's Generalized Failure Theory". The Tsai-Wu failure criterion is a simplification of this criterion. On the other hand, many different failure theories have been made in the past to improve those failure criteria [56-58] that do not suffer from shortages and are in a better fit for experimental results. These situations are still an open subject of composite material research and the sometimes passioned debate [59, 60] has yet to end with a complete conclusion.

The Tsai-Wu failure criterion is one of the most used composite failure criteria. The other preceding failure criteria that consider the biaxial failure have some disappointments. One obvious way to improve the fit between the criterion and experiment results is to increase the count of terms that are used in the prediction equation. This increase in curve-fitting capability and the added characteristics of representing the various strengths in tensor form were dwelled on by Tsai and Wu. [61] In this process, a new strength definition is required to represent the interaction between stresses in two directions [62].

2.10 Linear Elasticity Types & Quasi-Isotropic Laminates

Linear elasticity is a type of elasticity when elastic properties do not change under a stress load. Namely, Poisson's ratio and young's modulus is constant in the linear elasticity. Also, the shear modulus is also constant according to these constancies. There are 3 types of linear elasticities, those are isotropic elasticity, anisotropic elasticity and orthotropic elasticity. Isotropic elasticity is the elasticity type that has the same elastic properties in all directions. The anisotropic elasticity is the elasticity type that has different elastic properties in all directions. The orthotropic elasticity is the elasticity type that has different elastic properties in 3 directions (X, Y, and Z). Also, orthotropic elasticity can be defined as a simplified version of anisotropic elasticity. Also, the laminates can behave with different elasticities when they are stacked up at different angles. One of the most used stacking ways is quasi-isotropic stack-up. The plies of the laminate can be stacked up in such a way that the laminate will have an isotropic layer under in-plane loading. Actually, the laminate is orthotropic under transverse loading and under inter-laminar shear, its behavior is different from an isotropic layer [63].

2.11 The Main Applications of Sandwich Composite Structures in the Aerospace Industry

The difficulty of designing a structure as light as possible when not decreasing the stiffness and strength is a basis in the aerospace structure design. Namely, the requirement needs to stabilize thin panels used in the structure to resist tensile, torsional, bending and compressive loads. Traditional airframe structure design has in the past, and still does to some improvement, use stringers, ribs, or frames with longitudinal stiffeners and stabilizing rings to overcome this difficulty. However, this is not an incisive solution in composite design. Stabilization of a surface in order to provide resistance to deforming forces can be designed more efficiently by the use of two skin panels with a stabilizing and load-transferring medium panel between them [64].

Sandwich structure is a common principle in nature and hence the concept is older than a human being itself. The bone parts in the skeleton sections of animals and humans are sandwich structures with irregular type (foam) core materials. When we consider the bones in legs have to handle repetitive, combined bending and compression loads. Furthermore, nature force a strict demand for lightweight primary structures (e.g. skeletons of birds). All the mentioned examples show the main idea of structural optimization: minimum use of material to reduce the weight when providing efficient strength and performance [65].

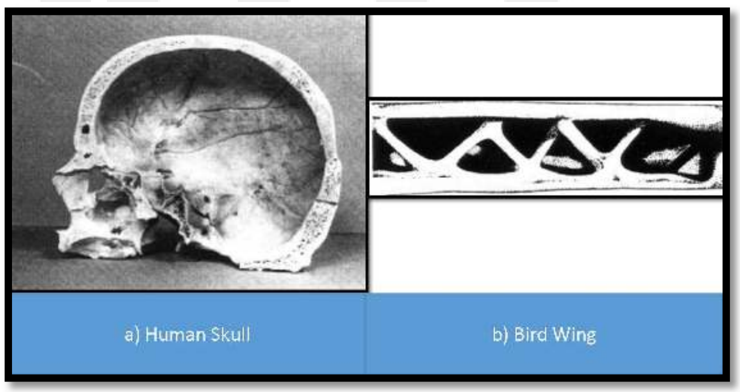

Figure 2.2 Natural Sandwich Composites

The aerospace applications of sandwich composites can be considered in 6 different sections. These are commercial aircraft applications, gas turbine engine applications, helicopter applications, fighter aircraft applications, spacecraft applications and satellite applications [66].

2.11.1 Commercial Aircraft Applications

Figure 2.3 Commercial Aircraft Applications

1. Radome: Specialized glass prepregs & flexcore honeycomb
2. Landing Gear Doors and Leg Fairings: Glass/carbon prepregs, honeycomb and Redux® bonded assembly
3. Galley, Wardrobes, Toilets: Fabricated Fibrelam panels
4. Partitions: Fibrelam panel materials
5. Wing to Body Fairing: Carbon/glass/aramid prepregs, honeycombs, Redux adhesive.
6. Wing Assembly: (Trailing Edge Shroud Box) Carbon/glass prepregs, Nomex honeycomb, Redux bonded assembly
7. Flying Control Surfaces - Ailerons, Spoilers, Vanes, and Flaps: Glass/carbon/aramid prepregs, honeycomb, Redux adhesive
8. Passenger Flooring: Fibrelam panels
9. Engine Nacelles and Thrust Reversers: Carbon/glass prepregs, Nomex honeycomb, special processed parts
10. Cargo Flooring: Fibrelam panels
11. Overhead Storage Bins: Prepregs/fabricated Fibrelam panels
12. Airstairs: Fabricated Fibrelam panels

2.11.2 Gas Turbine Engine Applications

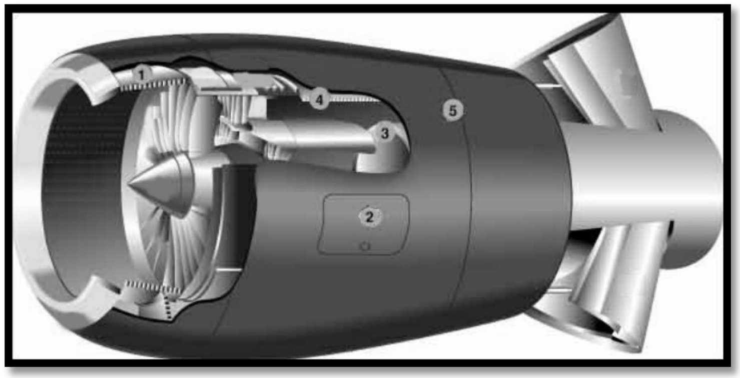

Figure 2.4 Gas Turbine Engine Applications

1. Acoustic Lining Panels: Carbon/glass prepregs, high temperature adhesives, aluminum honeycomb
2. Engine Access Doors: Woven and UD carbon/glass prepregs, honeycomb, and adhesives
3. Compressor Fairing: BMI/epoxy carbon prepreg, honeycomb and adhesives
4. Bypass Duct: Epoxy carbon prepreg, non-metallic honeycomb, and adhesives
5. Nacelle Cowling: Carbon/glass prepregs and honeycomb

2.11.3 Helicopter Applications

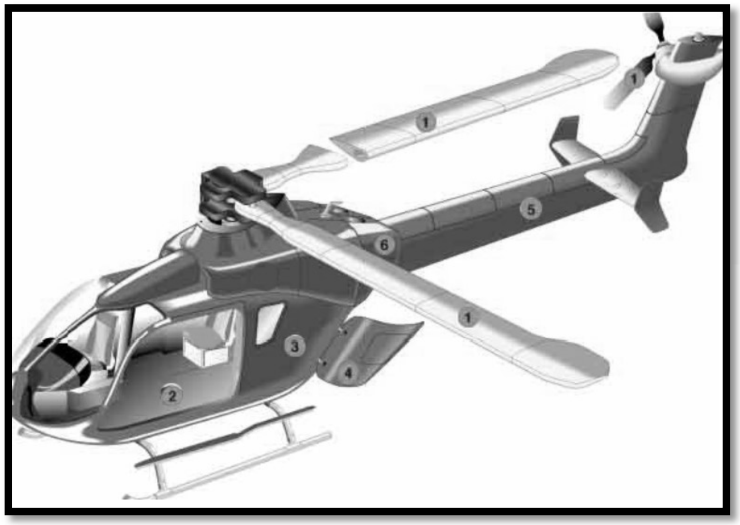

Figure 2.5 Helicopter Applications

1. Rotor Blades: Prepregs/carbon/glass honeycombs. Machined Nomex cores. Redux adhesives
2. Flooring: Fibrelam panels
3. Fuselage: Carbon and glass Prepregs. Honeycomb
4. Main and Cargo Doors: Epoxy carbon/glass prepreg, honeycomb and Redux adhesive
5. Boom and Tail Section: Epoxy carbon/glass prepreg, honeycomb and Redux adhesive
6. Fuselage Panels: Epoxy carbon/glass prepreg, Nomex honeycomb and Redux adhesives

2.11.4 Fighter Aircraft Applications

Figure 2.6 Fighter Aircraft Applications

1. Fuselage Panel Sections: Epoxy carbon prepregs, non-metallic honeycomb core and Redux adhesives
2. Flying Control Surfaces: Epoxy carbon and glass prepregs, honeycomb core material and Redux adhesives

2.11.5 Spacecraft Applications

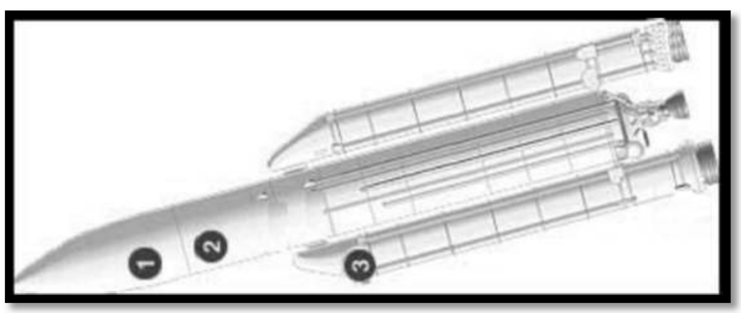

Figure 2.7 Spacecraft Applications

1. Fairings: Carbon prepregs, aluminum honeycomb and adhesives.
2. External Payload Carrier Assembly (SPELTRA): Carbon prepregs, aluminum honeycombs, and adhesives.
3. Booster Capotage: Epoxy glass/non-metallic honeycomb.

2.11.6 Satellite Applications

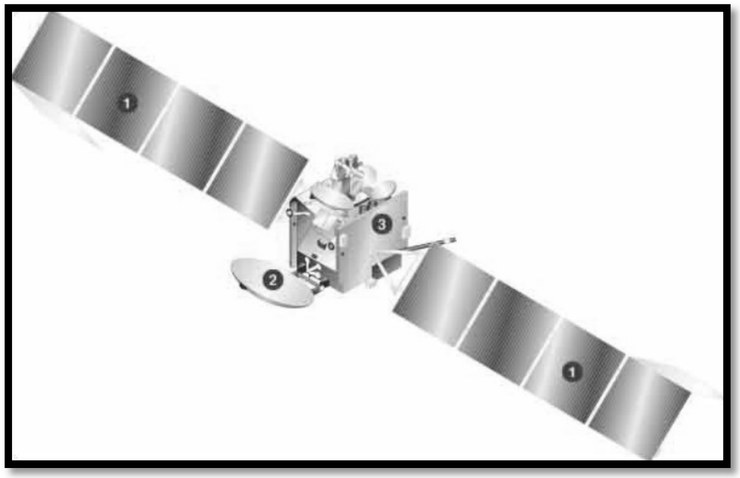

Figure 2.8 Satellite Applications

1. Solar Panels: Epoxy carbon prepregs, aluminum honeycomb, film adhesive
2. Reflectors Antennae: Epoxy/aramid prepreg, cyanate carbon prepreg, aramid/aluminum honeycomb
3. Satellite Structures: Carbon prepreg, aluminum honeycomb, film adhesive

CHAPTER 3

OPTIMIZATION ASSUMPTIONS AND PARAMETERS GENERATION

Since a parametric optimization depends on the parameters which are used in that optimization, the first step must be the proper selection of the parameters and their applications.

3.1 Optimization Assumptions

The optimization process considers some material and geometrical assumptions. These assumptions ease the process and pave the way for us to automatize the process.

3.1.1 Optimization Assumption of the Honeycombs

The honeycombs are produced by a combination of poly-aramid sheets. However, the thickness of the combined sheet in the combination edges is two times larger than the sheet thickness. This optimization is done by not considering the detailed geometry. The macroscopic element properties are used in order to ease the analyses for optimization since the global optimization matrix has a huge number of solution points. Some different validation cases are analyzed in order to see that we can obtain the same buckling load multiplier and IRF values. Those validation cases consider two situations one of them includes two times more ply number and the other case has 2 times thicker ply thickness.

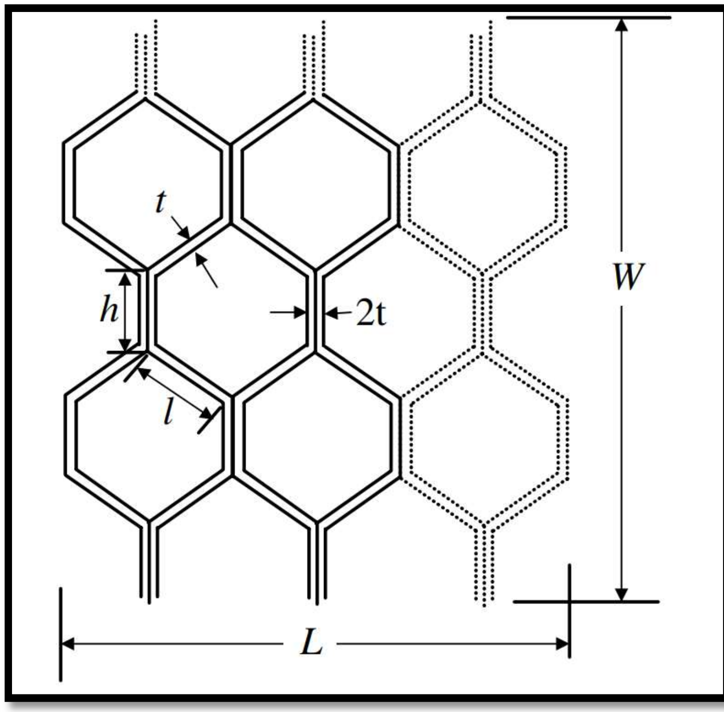

Figure 3.1 Honeycomb Sheet Connection Edges [67]

To consider this the honeycombs are considered in finite element analysis as orthotropic materials. The whole of the detailed geometry was not modeled. Honeycomb structures have orthotropic elasticity when considered from the macroscopic view. Honeycombs are assumed as continuous materials having orthotropic elasticity with low density. This assumption is necessary to have less mesh in the optimization. If the 3D model is used in the optimization, then, the mesh number and analysis time will increase abruptly as much as the solution files size increase.

3.1.2 Optimization Assumption of the Face Materials

The face materials are laminated composite materials, however, the thickness optimization with laminated composite is not possible when there is changing pre-preg number. For this case, all of the plies must be stacked up in the finite element analysis software to do optimization in this way. The parametrization with stacking cannot be done for a large number of parameters. Thus, lamination is assumed to have the same number of pre-preg fabrics. The pre-preg fabric thickness changing for optimization. The exceeded thicknesses of the pre-preg material is adapted to the suitable case in the post-process part of the analysis. The prepreg fabric is assumed to have the same resin and carbon fiber density for different thickness values.

3.2 Parameters Generation

This optimization considers the parameters of face panels and honeycomb core. Since these parameters change the macro-mechanical material properties such as elastic modulus in a reduced direction, the material properties must be defined according to these parameters in order to analyze the mathematical model generated with FEA.

3.2.1 Used Materials

This optimization was done by using 1 orthotropic uni-axial carbon-fiber prepreg and 1 orthotropic woven carbon-fiber prepreg for the face materials and isotropic aramid material for the honeycomb core. Isotropic aramid materials have macro-mechanical orthotropic characteristics when used in cellular solids. The honeycomb and the face materials are defined as orthotropic materials in FEA. The macroscopic elasticity of the honeycomb core changes with the cellular property. Even though the constitutive material has isotropic elasticity, the macroscopic elasticity of the honeycomb cores is orthotropic.

Two of the face material properties are obtained from the Ansys workbench material library. It is the same with the “Epoxy Carbon UD (395 GPa) Prepreg” and “Epoxy Carbon Woven (230 Gpa) Prepreg” material in the material library.

The honeycomb consists of polyaramid sheets. The material properties for optimization are found in the literature. The Poisson’s ratios for the found materials are missing. The Poisson’s ratio parameters are assumed the same as the honeycomb material in the Ansys Workbench material library. The honeycomb material properties found in 2 different books have been compared and the optimization material properties are chosen by this way.

Table 3.1 Ansys Workbench Default Honeycomb Material Properties

Property	Value	Unit
Density	80	kg/m^3
Young's Modulus X	1	Mpa
Young's Modulus Y	1	Mpa
Young's Modulus Z	255	Mpa
Poisson's Ratio XY	0.49	-
Poisson's Ratio YZ	0.001	-
Poisson's Ratio XZ	0.001	-
Shear Modulus XY	1	Pa
Shear Modulus YZ	37	Mpa
Shear Modulus XZ	70	Mpa

Table 3.2 Used Honeycomb Properties in the Source [68]

Cell Size	Density	Paper Thickness	Compressive Strength	L-Shear Strength	L-Shear Modulus	W-Shear Strength	W-Shear Modulus
inch	**lb/ft^3**	**inch**	**psi**	**psi**	**ksi**	**psi**	**ksi**
1/8	1.8	0.0015	131	87	3.9	44	1.7
1/8	3	0.0020	334	203	7	131	3.8
1/8	4	0.0020	595	290	9.1	174	5.5
3/16	1.8	0.0015	87	73	3.3	44	1.9
3/16	2	0.0020	160	102	4.4	58	2.3
3/16	3	0.0020	305	174	6.2	102	3.8
3/16	4	0.0020	522	247	8	160	5.1

1/4	1.5	0.0015	87	58	2.9	29	1.5
1/4	2	0.0015	145	102	3.9	58	2.2

Some of the honeycomb material properties are missing for the definition in the finite element analysis. Z modules need to be found. To find the Z modules of the honeycombs, **Equation 1.4** is useful. The ratio of macro-mechanical density is assumed the same as the ratio of Z modulus to the Nomex sheet modulus.

3.2.2 Face Material Stack-up Parameter Generation

Since the face material parametrization depends on the effective lamination, the angle densities must be decided for the materials. 0, +/-45, and 90-degree plies are used for the optimization. There are 4 arbitrarily decided effective laminated materials for face materials in the optimization.

Table 3.3 Face Materials Ply Densities

PLY ANGLE	**0**	**+/- 45**	**90**
Stack-up 1 % Ply Densities	25	50	25
Stack-up 2 % Ply Densities	60	20	20
Stack-up 3 % Ply Densities	40	40	20
Stack-up 4 % Ply Densities	20	40	40

The angle densities do not define the stacking. The stacking of the lamina is important for the composite materials. This stacking changes the mechanical properties. This change depends on angle orientation density and laminate stacking order. This thesis work considers 2 lamination types, those are symmetric and quasi-isotropic laminates. Christos Kassapoglou dwells on that there are several special types of lamina stacking. Some of them are symmetric, balanced, cross-ply, angle-ply, and quasi-isotropic lamination. All of the lamination types have different usage areas. When a laminate has a symmetric stacking sequence in the laminate, then, this is a symmetric laminate. This sequence is generated according to the laminate mid-plane. Namely, the material thickness and orientation of each pair of plies located symmetrically to the laminate across the middle symmetry plane in the same sequence. Quasi-isotropic laminates are special laminates that have the same stiffness in any direction in the lamination plane. Quasi-isotropic stacking can be done by considering creating a sequence of n plies. This sequence requires that there must be no direction that has more fibers than any other direction [69].

The optimization model calculation time will be exaggerated when the optimization parameters are defined by hand for all models. To reduce the solution time and have a realizable optimization process, the calculations are done for constant stacking orientation and changing fiber thickness. There is 4 constant stacking orientation for 4 different densities. The stacking orientation considers the fiber angles and lamination in those directions.

Different stacking of composite materials can have the same macro-mechanical elasticity parameters. The following 3 laminate orientations $[0/45/90/-45]_S$, $[45/-45/0/90]_S$, and $[45/0/-45/90]_S$ have the same mechanical properties [70]. To have 4 suitable stacking the lamination angle orientation must be considered.

3.2.2.1 Stack-up for [25/50/25] Ply Density

The selected lamination array: $[45/90/-45/0]_S$

3.2.2.2 Stack-up for [60/20/20] Ply Density

The selected lamination array: $[0/0/0/-45/90/90/45/0/0/0]_S$

3.2.2.3 Stack-up for [40/40/20] Ply Density

The selected lamination array: $[90/45/0/-45/0]_S$

3.2.2.4 Stack-up for [20/40/40] Ply Density

The selected lamination array: $[90/-45/90/45/0]_S$

Where the “S” is the symmetry plane of the composite panel. The lamination angles and densities are symmetric across the symmetry plane.

3.2.3 Face Material Fabric Thickness Parameter Generation

To have a realizable optimization process. The lamination process is reduced into 4 constant stack-up, namely, the ply number and fiber orientation assumed the same for one of that 4 stack-up. The lamination is done with uni-directional carbon fiber prepreg materials. The prepreg material thickness is decided with the optimization process. The optimization thickness is defined as continuous thickness for optimization. The post-process of the optimization is used to decide which thickness is the best for the optimization. 8 different thickness values are selected from a manufacturer’s material table whose name is “Hitex”.

Table 3.4 Stock Prepreg Thicknesses [71]

Thickness Number	Unit [mm]
Thickness 1	0.08
Thickness 2	0.10
Thickness 3	0.13
Thickness 4	0.15
Thickness 5	0.18
Thickness 6	0.20
Thickness 7	0.23
Thickness 8	0.25

The optimization process decides which thickness parameter is optimum for the production with post-processing calculation. This thickness parameter was added into analysis creating different fabrics which have the same material property and different thicknesses. Furthermore, to reach a useful optimized geometry, not only input parameters but also output parameters are used. The load multiplayer of the first mode from the eigenvalue buckling module, the maximum Tsai-Wu inverse reserve factor and the weight of the panel are used as output parameters in the optimization process.

3.3 Optimization Process

The global matrix for this multi-objective design is a very large. In order to reduce the number of design points to be analyzed, an optimization process was performed. This optimization has large numbers of dependent and variables. Therefore, optimization tools may converge to local minimum weight points. In order to prevent this, the global optimization matrix is created in an excel sheet and copied to analysis software. Analysis were completed with four optimization matrices. Three of them are for global search and one of them is for local search. The design logic can be seen in the **Figure 3.2**:

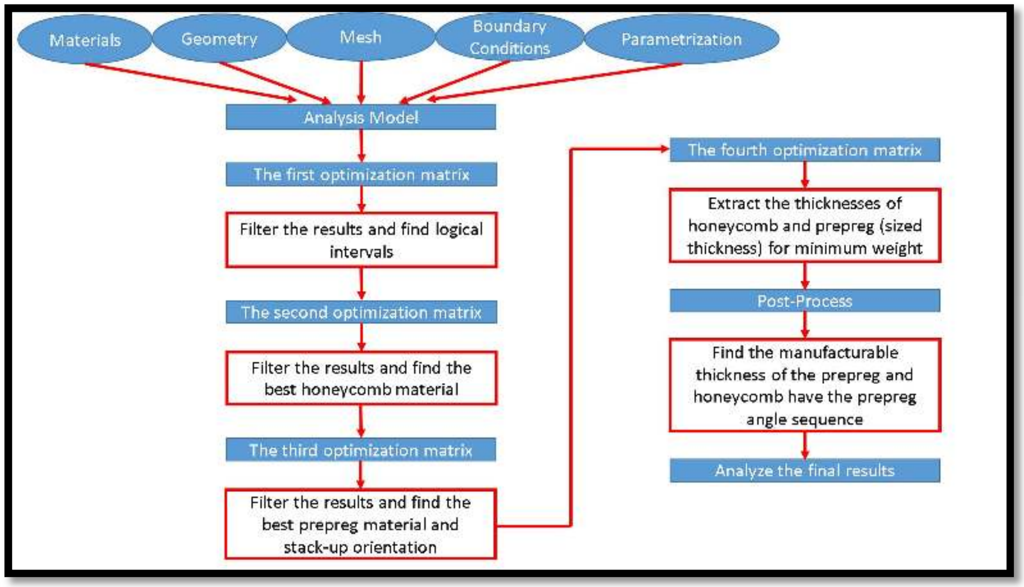

Figure 3.2 Schematic Representation of Optimization Logic

3.3.1 The First Optimization Matrix

The first optimization matrix had 72 local regions. These local regions were defined according to material and stack up combinations. 9 different honeycomb, 4 different stack-up, 2 different prepreg material combination creates 72 different regions. In this matrix, the prepreg thickness interval is 0.05 [mm] to 5 [mm] and honeycomb thickness interval is 50 [mm] to 500 [mm]. 100 design points for all local regions were generated. Then, the extracted results were filtered according to the IRF and buckling load multiplier. For the material combinations, logical intervals were found.

3.3.2 The Second Optimization Matrix

The second optimization matrix was prepared to find the best honeycomb material. For this purpose, the intervals that are extracted from the first optimization matrix result were used. The prepreg material, stack-up and thickness were constant in this second matrix. The honeycomb material and thickness were the variable parameters. After the solution, the results were filtered according to the minimum mass and maximum stiffness. Then the honeycomb material was selected.

3.3.3 The Third Optimization Matrix

The third optimization matrix was prepared to find the best prepreg material and stack-up orientation. In this optimization matrix, the intervals that had been extracted from the first optimization matrix result was used. While the honeycomb material and thickness were constant in this third matrix, the prepreg material, thickness and stack-up orientations were varied. After the solution, the results were filtered according to the minimum mass and maximum stiffness. Then the prepreg material and stack-up orientations were selected.

3.3.4 The Fourth Optimization Matrix

The fourth optimization matrix was prepared to find the best thickness for the materials. In order to do this, the intervals that had been extracted from the first optimization matrix result were used. However, the values were very sensitive in this matrix. The thickness sensitivity was 0.0001[mm] for the prepreg and 0.01 [mm] for the honeycomb. The honeycomb material, prepreg material and stack-up orientation were constant in this fourth matrix. The prepreg thickness and honeycomb thickness were the variables. After the solution, the results were filtered according to the minimum mass and maximum stiffness. Then the prepreg sizing thickness and honeycomb thickness were selected. The prepreg thickness and honeycomb thickness obtained from this optimization were not taken as the final value. The post-process section is for manufacturability. After the post-process, which considered manufacturability, the thicknesses were decided according to the availability of the stock materials.

3.4 Optimization Tool Validation

The optimization process was completed with the ANSYS Workbench FEA software. In order to validate the optimization tool, literature has been considered. The failure mode and buckling loads analysis that were completed with 2D shell elements in Ansys FEA software were validated according to the work of Turvey and Zhang [ref]. They claimed that the failure criteria results of the Tsai-Wu criterion can vary -4/+6 %, and buckling load can vary -6/+7 % from the test results [72]. This work is similar to our case when compared with the mesh type for composite material used in the analyses. Also, static failure and buckling load multiplier extraction were completed with the same solver. This case validates that Ansys software usage in the analysis is a reasonable selection.

3.5 Calculation of Missing Elastic Properties of Honeycombs

Honeycomb materials given in **Table 3.2** have some missing properties. In order to determine the missing properties to be used in the finite element analysis, the missing properties are calculated by exact formulas [73]. These formulas are used for the approximation of the commercial honeycombs' in-plane and out-of-plane mechanical properties. However, in the finite element analysis, the inverse Poisson ratio terms are not included. These values are assumed as same with the default honeycomb material in the library.

In-plane Formulas

$$\frac{E_x}{E_{sheet}} = \left(\frac{t}{l}\right)^3 \frac{\cos\theta}{\left(\frac{h}{l} + \sin\theta\right)\sin\theta^2} \tag{3.1}$$

$$\frac{E_y}{E_{sheet}} = \left(\frac{t}{l}\right)^3 \frac{\left(\frac{h}{l} + \sin\theta\right)}{\cos\theta^3} \tag{3.2}$$

$$\nu_{xy} = \frac{\cos\theta^2}{\left(\frac{h}{l} + \sin\theta\right)\sin\theta} \tag{3.3}$$

$$\nu_{yx} = \frac{\left(\frac{h}{l} + \sin\theta\right)\sin\theta}{\cos\theta^2} \tag{3.4}$$

$$\frac{G_{xy}}{E_{sheet}} = \frac{\left(\frac{h}{l} + \sin\theta\right)\sin\theta}{\cos\theta^2} \tag{3.5}$$

Out-of-plane Formulas

$$\frac{E_z}{E_{sheet}} = \left(\frac{t}{l}\right)\frac{\left(1+\frac{h}{l}\right)}{\left(\frac{h}{l}+\sin\theta\right)\cos\theta} \tag{3.6}$$

$$\nu_{zx} = \nu_{zy} = \nu_{sheet} \tag{3.7}$$

$$\nu_{xz} = \nu_{yz} \approx 0 \tag{3.8}$$

$$\frac{G_{sheet}}{E_{sheet}} = \frac{1}{2(1+\nu_{sheet})} \tag{3.9}$$

$$\frac{G_{xz}}{G_{sheet}} = \left(\frac{t}{l}\right)\frac{\cos\theta}{\left(\frac{h}{l}+\sin\theta\right)} \tag{3.10}$$

$$\begin{aligned} G_{yz} = {} & \left(G_{she} \quad \left(\frac{t}{l}\right)\frac{\left(\frac{h}{l}+\sin\theta\right)}{\left(\frac{h}{l}+1\right)\cos\theta}\right) + \left(\frac{0.787 * (G_{sheet})}{\left(\frac{b}{l}\right)}\right) \\ & * \left(\left(\frac{t}{l}\right)\frac{\left(\frac{h}{l}+sin^2\,\theta\right)}{\left(\frac{h}{l}+\sin\theta\right)\cos\theta} - \left(\frac{t}{l}\right)\frac{\left(\frac{h}{l}+\sin\theta\right)}{\left(\frac{h}{l}+1\right)\cos\theta}\right) \end{aligned} \tag{3.11}$$

The calculated and inserted properties can be seen in **Table 4.6**; the elastic limits are not defined for the honeycombs.

CHAPTER 4

ANALYSIS SETUP & RESULTS

4.1 Definition of Materials in Workbench

The materials used in the analysis are defined in the workbench and saved in the software library as the first analysis step. There are 11 materials in the optimization analysis 1 of them is a metallic material, one of them is pre-preg material and 9 of them are honeycomb materials. The face panel optimization depends on stack-up and material thickness. The thesis work is using different materials, stack-ups and combinations. This generates a large optimization matrix to solve. For more parameters and their combinations, higher computation costs and time are needed. No need for that much complexity in this stage. Default material properties of prepregs and metallic materials are obtained from Ansys Workbench material library [74].

Table 4.1 Default Structural Steel Properties

PROPERTY	VALUE	UNIT
Density	7850	$\frac{kg}{m^3}$
Young's Modulus	200	Gpa
Poisson's Ratio	0.3	-

Table 4.2 Default Titanium Alloy Properties

PROPERTY	VALUE	UNIT
Density	4620	$\frac{kg}{m^3}$
Young’s Modulus	96	Gpa
Poisson’s Ratio	0.36	-

Table 4.3 Default Aluminum Alloy Properties

PROPERTY	VALUE	UNIT
Density	2770	$\frac{kg}{m^3}$
Young’s Modulus	71	Gpa
Poisson’s Ratio	0.33	-

Table 4.4 Default 395 Gpa Uni-Directional Prepreg Properties

PROPERTY	VALUE	UNIT
Density	1540	$\frac{kg}{m^3}$
Young’s Modulus X	209	Gpa
Young’s Modulus Y	9.45	Gpa
Young’s Modulus Z	9.45	Mpa
Poisson’s Ratio XY	0.27	-
Poisson’s Ratio YZ	0.4	-
Poisson’s Ratio XZ	0.27	-
Shear Modulus XY	5.5	Gpa
Shear Modulus YZ	3.9	Gpa
Shear Modulus XZ	5.5	Gpa
Tensile Stress Limit X	1979	Mpa

Tensile Stress Limit Y	26	Mpa
Tensile Stress Limit Z	26	Mpa
Compressive Stress Limit X	-893	Mpa
Compressive Stress Limit Y	-139	Mpa
Compressive Stress Limit Z	-139	Mpa
Shear Stress Limit XY	100	Mpa
Shear Stress Limit YZ	50	Mpa
Shear Stress Limit XZ	100	Mpa
Tensile Strain Limit X	0.0092	-
Tensile Strain Limit Y	0.0031	-
Tensile Strain Limit Z	0.0031	-
Compressive Strain Limit X	-0.0053	-
Compressive Strain Limit Y	-0.0172	-
Compressive Strain Limit Z	-0.0172	-
Shear Strain Limit XY	0.016	-
Shear Strain Limit YZ	0.014	-
Shear Strain Limit XZ	0.016	-
Tsai-Wu Coupling Coefficient XY	-1	-
Tsai-Wu Coupling Coefficient YZ	-1	-
Tsai-Wu Coupling Coefficient XZ	-1	-

Table 4.5 Default 230 Gpa Woven Prepreg Properties

PROPERTY	VALUE	UNIT
Density	1420	$\frac{kg}{m^3}$
Young's Modulus X	61.34	Gpa
Young's Modulus Y	61.34	Gpa
Young's Modulus Z	6.9	Mpa

Poisson’s Ratio XY	0.04	-
Poisson’s Ratio YZ	0.3	-
Poisson’s Ratio XZ	0.3	-
Shear Modulus XY	3.3	Gpa
Shear Modulus YZ	2.7	Gpa
Shear Modulus XZ	2.7	Gpa
Tensile Stress Limit X	805	Mpa
Tensile Stress Limit Y	805	Mpa
Tensile Stress Limit Z	50	Mpa
Compressive Stress Limit X	-509	Mpa
Compressive Stress Limit Y	-509	Mpa
Compressive Stress Limit Z	-170	Mpa
Shear Stress Limit XY	125	Mpa
Shear Stress Limit YZ	65	Mpa
Shear Stress Limit XZ	65	Mpa
Tensile Strain Limit X	0.0126	-
Tensile Strain Limit Y	0.0126	-
Tensile Strain Limit Z	0.008	-
Compressive Strain Limit X	-0.0102	-
Compressive Strain Limit Y	-0.0102	-
Compressive Strain Limit Z	-0.012	-
Shear Strain Limit XY	0.022	-
Shear Strain Limit YZ	0.019	-
Shear Strain Limit XZ	0.019	-
Tsai-Wu Coupling Coefficient XY	-1	-
Tsai-Wu Coupling Coefficient YZ	-1	-
Tsai-Wu Coupling Coefficient XZ	-1	-

Table 4.6 Honeycomb 1-9 Orthotropic Properties

PROPERTY	1	2	3	4	5	6	7	8	9	UNIT
Density	1.8	3	4	1.8	2	3	4	1.5	2	$\frac{lb}{ft^3}$
Young's Modulus X	3900	7000	9100	3300	4400	6200	8000	2900	3900	psi
Young's Modulus Y	1700	3800	5500	1900	2300	3800	5100	1500	2200	psi
Young's Modulus Z	13300	22200	29600	13300	14800	22200	29600	11100	14800	psi
Poisson's Ratio XY	0.49	0.49	0.49	0.49	0.49	0.49	0.49	0.49	0.49	-
Poisson's Ratio YZ	0.001	0.001	0.001	0.001	0.001	0.001	0.001	0.001	0.001	-
Poisson's Ratio XZ	0.001	0.001	0.001	0.001	0.001	0.001	0.001	0.001	0.001	-
Shear Modulus XY	1	1	1	1	1	1	1	1	1	Pa
Shear Modulus YZ	1900	3200	4300	1900	2100	3200	4300	1600	2100	psi
Shear Modulus XZ	3700	6100	8100	3700	4100	6100	8100	3000	4100	psi

4.2 Analysis Geometry

Analysis geometry is derived from an arched upper panel of the wing described in the **Figure 4.1**. A shell panel and 2 lines are used to represent the geometry from an arbitrary rib to the next rib. The shell panel is optimization geometry and the line bodies are used for preventing tip buckling as they represent the rib stiffness.

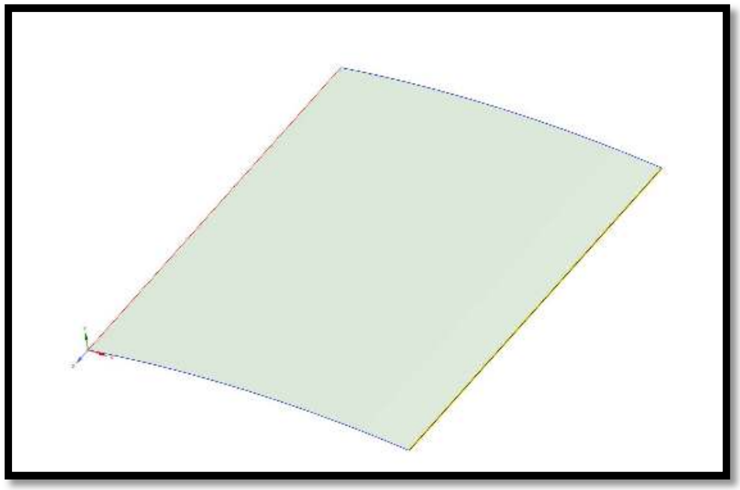

Figure 4.1 Used Geometry in the Analyses

The blue edges are line bodies, and the red edge and the yellow edge are free edges. The panel length is 2500 [mm], panel the width is 2000 [mm], and the arch angle is 15 degrees. Also, the line bodies are defined as default structural steel material and simple bars having a 30 [mm] radius circular cross-section.

4.3 Mesh Quality & Number of Nodes

The optimization analysis needs different modules of the Ansys Workbench software. These modules are necessary for creating material, creating parameters and solving solid mechanics problems. To have good results, mesh quality is important.

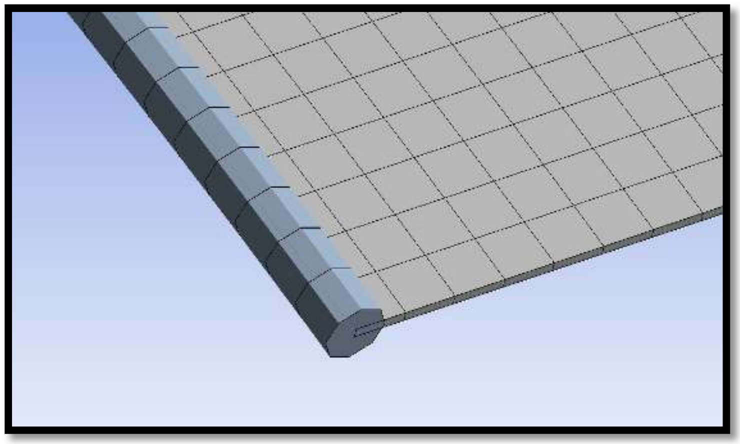

Figure 4.2 Mesh View 1

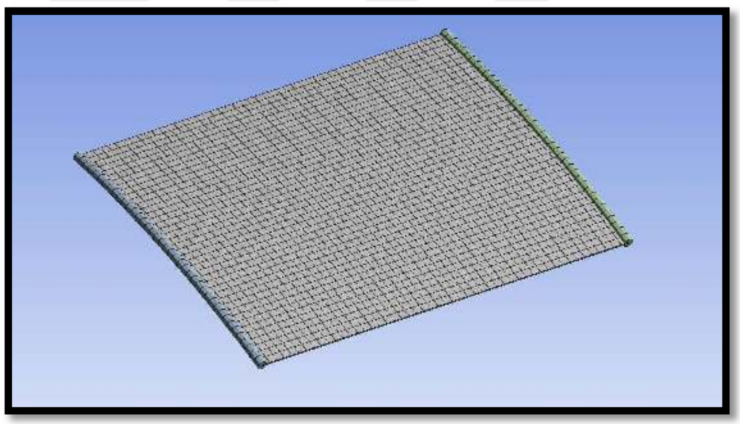

Figure 4.3 Mesh View 2

Table 4.7 Mesh Quality Deviation

Property	Element Quality
Minimum	0.98659
Maximum	0.99947
Average	0.99853

Since the geometries are simple, the average mesh quality is very close to the perfect mesh quality which is "1". This is necessary to have a perfectly completed optimization.

Table 4.8 Mesh Element & Node Number

Mesh Content	Number
Node	2171
Element	2080

In the optimization, 2171 nodes and 2080 linear elements are used. Since the used elements are 1D and 2D elements, the number of the elements are relatively lower than a 3D FEA. Optimization steps are solved around 6 seconds and with this method a lot of combination results are solved stored.

4.4 Boundary Conditions

Multi load conditions are used for the optimization. The load on the body can be expressed as axial compressive load, vertical shear load and plane shear load. The plane shear load also paves the way for that optimization can select fiber orientation density of 45 degrees. Otherwise, the lowest density stack-up will be selected. Since the plane shear load is carried by 45 degree oriented fibers, there must be some 45 degree oriented fibers in the stack-up.

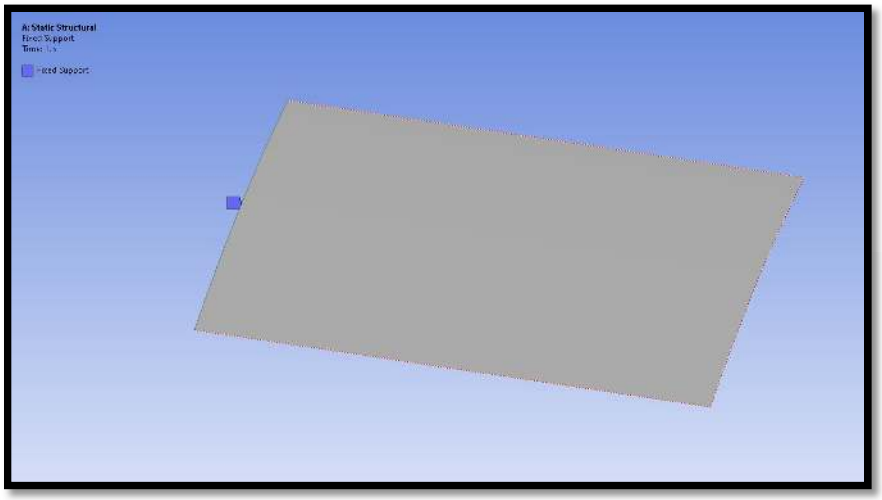

Figure 4.4 Fixed Support

Fixed support restricts the movement in all degrees of freedom of the nodes.

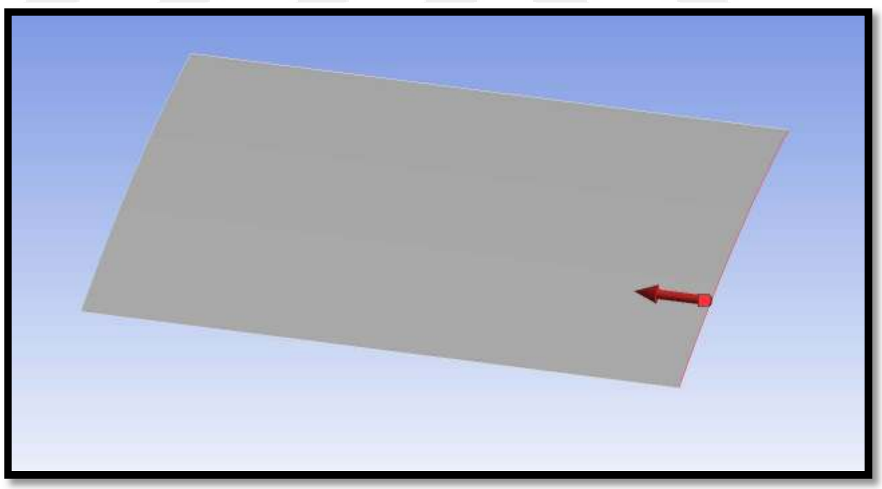

Figure 4.5 Axial Load

The axial load is 200000 [N]

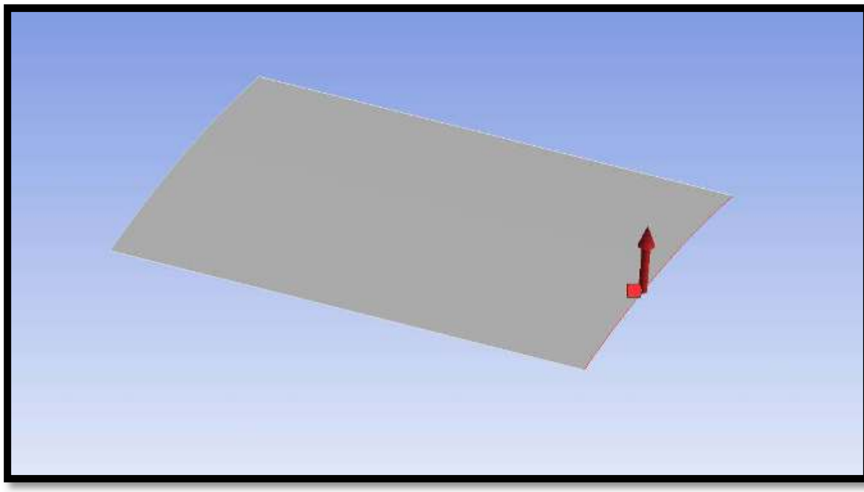

Figure 4.6 Vertical Shear Load

The vertical shear load is 120000 [N]

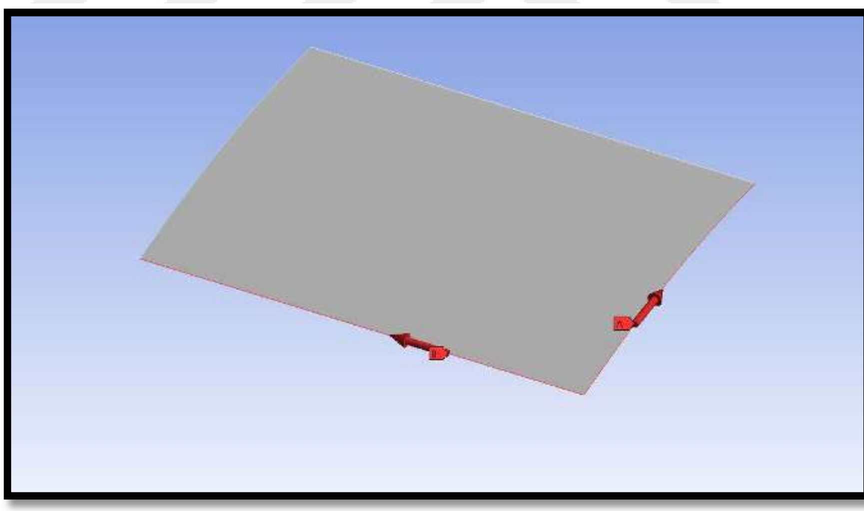

Figure 4.7 Plane Shear Load

The plane shear load is 500000 [N]. They are applied with two forces on 2 edges with 2 forces having the same value of 500000 [N] magnitude in different plane shear directions.

4.5 Analysis Setup and Definition of the Parameters in the Analysis

This chapter is the most important part of the analysis. The optimization process uses different parameters from different modules of the software.

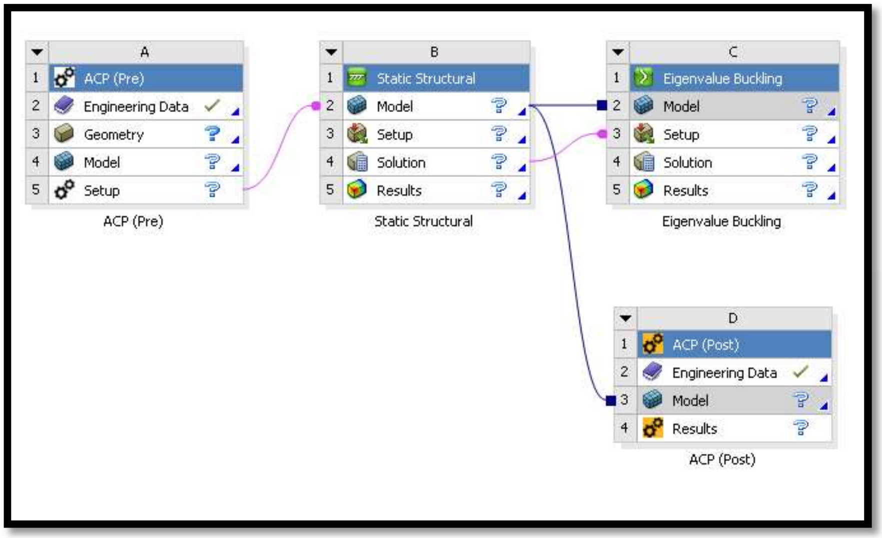

Figure 4.8 Workbench Modules & Orientations

The first module named “ACP (Pre)” which is in the “A” box in the **Figure 4.8** is used for composite material and its parameters. The second module named “Static Structural” which is in the “B” box in the **Figure 4.8** is used for static results that are needed for extraction of buckling load multiplier mode. The third module named “Eigenvalue Buckling” which is in the “C” box in the **Figure 4.8** is used for extraction of load multiplier for the first mode. The last module named “ACP (Post)” which is in the “D” box in the **Figure 4.8** is used for weight parameters as out parameters.

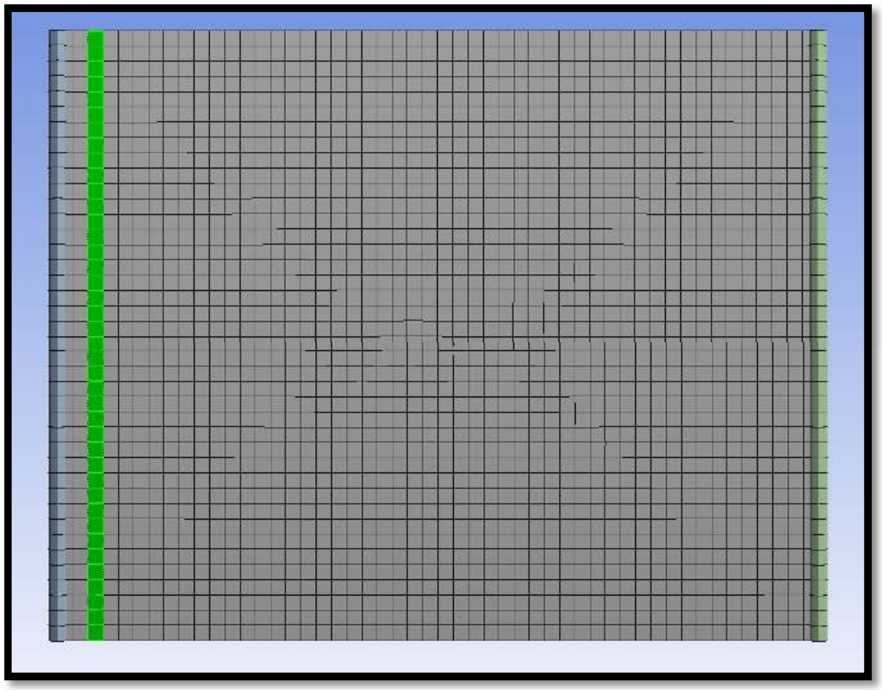

Figure 4.9 The Elements Used for Tsai-Wu Inverse Reserve Factor

The 2 inter-analysis parameters that change between analysis iterations are extracted from the “Eigenvalue Buckling” and “ACP-(Post)” modules. To find the first mode of load multiplier that is the first inter-analysis parameter extracted from the buckling module, the negative load multiplier solutions are closed. The safety margin for the first load multiplier is accepted as “0.01” for the first load multiplier. That is the first inter-analysis parameter. Second one is Tsai-Wu inverse reserve factor for the plies are added into the parameters with a limit value of 0.5. The meshes are selected 2 before from the fixed support since the maximum Tsai-Wu failure probe is extracted from singularity eliminated meshes. Optimization is defined for keeping the first load multiplier above “1.01” and keeping the inverse reserve factor below 0.5. The weight that is the second inter-analysis parameter is extracted from the “ACP post” module is defined in the optimization for reducing that parameter. To decrease the optimization time, the honeycomb material parameter was selected by first parametric analysis. The stiffness/weight ratio is used to select the best-fitted honeycomb material in 9 defined materials. The stiffness/weight ratio of the “honeycomb 3” material and “honeycomb 7” material have the same and highest value between 9 materials.

Table of Design Points

	A	B	C	D	E	F
1	Name	P1 - prepregfabric.thickness	P2 - honeycomb.material	P3 - honeycomb.thickness	P4 - upperpanel.ply_material	P5 - lowerpanel.ply_material
2	DP 0 (Current)	0.0005	1	0.002	1	1
3	DP 1	0.002	2	0.003	3	1
4	DP 2	0.002	8	0.004	4	3
5	DP 3	0.002	4	0.005	1	1
6	DP 4	0.002	4	0.01	1	1

Figure 4.10 A Part of Design Point Input Parameters Used in Parametric Optimization

G	H	I
P6 - Total Deformation Load Multiplier	P7 - Sensor.1.weight	P8 - Inverse Reserve Factor Maximum
		5.7012

Figure 4.11 A Part of Design Point Output Parameters Used in Parametric Optimization

The parametric optimization setup is prepared with different combinations of materials and thicknesses. The lightest design point is selected from these combinations.

4.6 Analysis Results

Analysis results of more than 12000 design points are obtained and the most suitable design point is selected. The design point has first load multiplayer more than 1.01 and Tsai-Wu inverse reserve factor less than 0.5. According to analysis results, the optimized combination selects the epoxy woven 230 [Gpa] prepreg as face prepreg material, 2nd stack-up combination as face panel material, the 3rd honeycomb material as honeycomb core material. The optimized pre-preg thickness is 0.3164 [mm], optimized honeycomb thickness is 170.78 [mm]. The optimized material and geometrical properties have a 5.65 value as load multiplayer and the inverse reserve factor value is 0.50 according to the Tsai-Wu failure criterion with 144.98 [kg] as the total weight honeycomb core and face materials.

Table 4.9 Analysis Response Result Table (Before Post-Process)

	Prepreg Material	Honeycomb Material	Prepreg Thickness	Honeycomb Thickness	Stack-up Orientation
Optimization Response	Woven 230 Gpa Prepreg	3	0.3164 [mm]	170.78 [mm]	2

4.7 Analysis Post-Process

The optimization process was not a complete design, thus, a post-process was necessary to have a better methodology. This post-process section is necessary for 2 main reasons. The first reason is making the optimization suitable for production with stock materials. The second reason is to make the optimization suitable for the rules of thumb of the composite material design. Some of these rules dwelled on by Christos Kassapoglou are listed below [75]:

4.7.1 Stacking Sequence Related Rules

- The stacking sequence of a laminate should be symmetric to eliminate unwanted membrane/bending coupling (B matrix = 0).
- The stack-up should be balanced for every +θ ply there should be a –θ ply of the same material and thickness somewhere in the laminate to eliminate stretching/shearing coupling ($A_{16} = A_{26} = 0$).
- Bending/twisting coupling should be avoided.
- At least 10% of the fibres in every layup should be lined up with each of the four principal directions: 0°, 45°, −45°, and 90°. This protects against secondary load cases.
- The number of unidirectional plies with same orientation next to each other must be minimized.

4.7.2 Loading and Performance Related Rules

- To improve the bending stiffness of a one-dimensional composite structure, place 0° plies as far away from the neutral axes as possible.
- Place +/- 45° plies as far away from the neutral axis as possible in order to improve buckling and crippling resistance.

- To improve damage resistance, +/- 45°fabric plies should be placed on the outside of a stacking sequence.
- Skin layup should be dominated by +/- 45° plies in order to improve the shear stiffness and strength of the layup.
- External plydrops must be avoided. Plydrops should be as close to the mid-plane of the laminate as possible.

4.7.3 Environmental Sensitivity and Manufacturing Related Rules

- For lightly loaded structures, the thickness should be no lower than 0.5–0.6 mm to keep moisture from seeping into the structure. For lower thicknesses, additional coating protecting against moisture should be used.
- Avoid 90° plies going around corners in particular when convex tools are used during layup. It is very hard to make the stiff 90° fibres. This is called as bridging and must be avoided.

4.7.4 Configuration and Layout Related Rules

- Although the structure is lightly loaded, stiffeners with a one-sided flange (L, C, Z) on the skin side should be avoided and stiffeners with flanges on either side of the web should be preferred (T, I, J, Hat). This protects the resin pocket present at the web flange corner from moisture and contamination and minimizes the possibility that matrix cracks may develop there and coalesce into delaminations under fatigue loading.
- Efficient stiffener and frame spacing lengths are similar to metal structures. The frame spacing is about 500- 510 [mm] and stiffener spacing is about 150-160 [mm].

The optimized panel have 144.98 [kg] mass in total. The optimized prepreg thickness result is 0.3164 [mm]. However, stock prepreg thickness does not have a value like this. The prepreg selection depends on having integer multiplayer with stock prepreg thickness. The prepreg fabric having 0.08 [mm] in **Table 3.5** is selected. The integer multiplayer is 4 for this lamination. 4 times more of the thickness value of the prepreg must lie above the optimization thickness and it is 0.32 [mm]. The optimized honeycomb thickness is 170.78 [mm], however, the production sensitivity is hard to apply and process validation is hard for 0.01[mm] or lower thickness levels. Thus, the honeycomb thickness is selected as 170.8 [mm] to stay above the optimization thickness. This stack-up has an array of $[0/0/0/\text{-}45/90/90/45/0/0/0]_S$ in the analyses. To clarify the results, analysis is completed with the optimized geometry using 0.08 [mm] prepreg and 178.8 [mm] honeycomb thickness with stack-up 2 face material combination and honeycomb 3 material. The prepreg angle orientation and numbers is changed due to the selection of the array. The angle orientation is $[0/\text{-}45/0/0/90/0/90/0/45/0/0/\text{-}45/0/0/90/0/90/0/45/0/0/\text{-}45/0/0/90/0/90/0/45/0/0/\text{-}45/0/0/90/0/90/0/45/0]_S$ after the post process. This orientation is selected and the array is defined in order not to stack up more than 3 prepregs in the same direction. The stack-up change affects the results minimally after the post-process. Tsai-Wu failure criteria results are affected by % 0.4 level. The lamination stack-up changes the result a bit higher. The post-processed buckling load multiplier is 6.10 whereas the sized geometry without lamination post-process is 5.69.

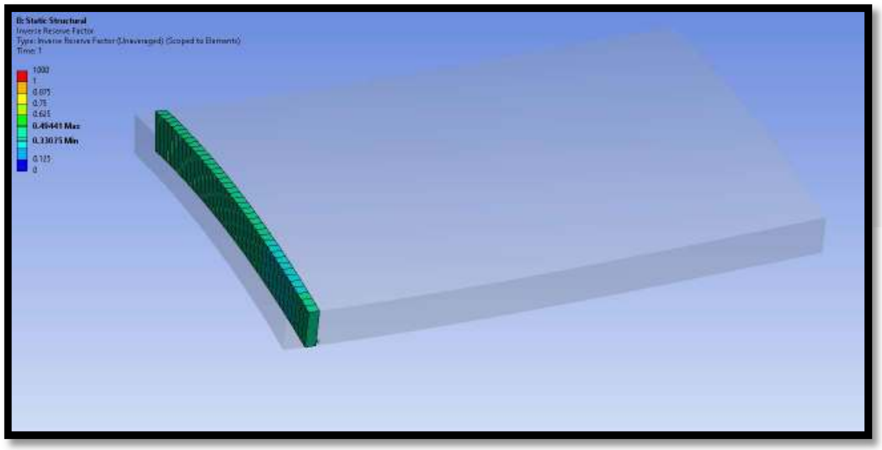

Figure 4.12 Optimized Geometry Inverse Reserve Factor on Critical Meshes

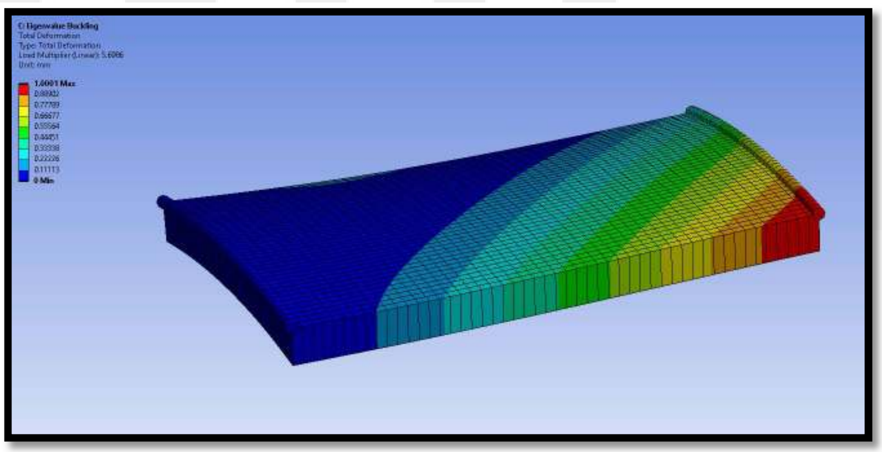

Figure 4.13 Optimized Geometry Buckling Load Multiplayer

As seen in the **Figure 4.12, 4.13**, the minimum buckling load multiplayer is 5.6986 and the maximum inverse reserve factor on the selected meshes according to Tsai-Wu failure criteria is 0.50.

Table 4.10 Analysis Response Result Table (After Post Processing)

Prepreg Material	**Honeycomb Material**	**Prepreg Thickness**	**Honeycomb Thickness**	**% Angle Densities (0,+/-45,90)**	**Prepreg Count Per Stack-up**
Woven 230 Gpa Prepreg	3	0.08 [mm]	170.80 [mm]	60/20/20	4

4.8 Weight Comparison with 3 Metal Alloys

The optimization step is repeated by considering the panel is made of 3 different and most used metal alloys in the aerospace industry. Those are steel, titanium and aluminum alloys. The material properties are obtained from the Ansys Workbench material library. The optimizations are completed by using the weight of the panel as a minimization objective and the buckling load multiplayer must be greater than 1.01. Then the optimization gives us different thickness and weight properties for different materials.

Table 4.11 Mass Comparison & % Difference

Material	Metal Panel Mass [kg]	Sandwich Panel Mass [kg]	% Difference
Structural Steel	775.50	144.98	81.30
Titanium Alloy	649.70	144.98	77.69
Aluminum Alloy	451.55	144.98	67.89

The optimization has reduced the mass when compared with the most used 3 different metal alloys. Although steel and titanium materials are not used in the skin geometries in the wings, the sizing and comparison is completed. This comparison shows us that sandwich composite material pave the way for that the design can be much lighter in airframes with composite materials. Thus, this is necessary and we can the now decide the combinations of materials that creates the optimum sandwich composite material with this method.

CHAPTER 5

CONCLUSIONS AND FUTURE WORK

The optimization for the sandwich composite panel was completed and the weight of the sandwich composite material was reduced. The best combination founded by optimization response in terms of prepreg material, honeycomb material, prepreg thickness shows us that the optimization with assumptions has a great advantage. The full model analysis has nearly the same results as the analyses done with assumptions. The differences occur by the post-process, however, it is necessary for the production limits. Furthermore, the finite element analysis considers the macro-mechanical behavior of this orientation in sandwich composite materials. However, the micro-mechanical property may create some unforeseen failures with FEA results due to production such as a break in the resin. The fabric orientation might be optimized in order to minimize the fabrics that are adjacent to each other and have the same angle of orientation as another post-process. This developed method paves the way for engineers to optimize and size the geometries much more effectively than the older methods done by using effective young modulus. Moreover, this optimization process considers and decides the prepreg material, stack-up, honeycomb material, honeycomb thickness and prepreg thickness. This is an improved methodology when compared with using classical lamination theory parameters that are effective young's modulus and shear modulus. Not only buckling load multiplayer but also failure criteria can be extracted when the sizing operation is being done.

This further optimization has 3 main advantages on classical lamination theory sizing done by using effective Young's modulus.

1) Material selection for the face material prepreg fabric and honeycomb core can be done with this method.

2) Buckling and static failure can be seen and there will be no future work occurred by failure since the optimization parameter depends on no failure condition.

3) Useful prepreg fabric thickness can be selected.

On the other hand, this method needs much more computational costs. Organizational and relatively big aerospace companies can handle this problem by using more separated licenses for the analyses. Thus, a simple airframe geometry with one load condition is created and modeled in the analyses instead of a full wing geometry with different load conditions. Also, restricted numbers of honeycomb material, pre-preg material, prepreg angle, and stack-up orientation are used in the optimization to decide, increasing this number will also generate a much larger optimization matrix. Considering much more parameters generates computational cost. Provided that a large number of software licenses with an HPC cluster, it is possible to optimize a whole airframe in terms of deciding component-based material selection, thickness, and orientation with this method.

The composite panel optimization process can be improved with the help of the usage of variable stiffness composite panels. Variable stiffened composite panel decreases the mass by using different thicknesses on the panel. The variable stiffness composite panel optimization also is a new and improving area for the composite structures. There are studies on the optimization for variable stiffness panels by using Bézier curves and intuitive fiber path definition methods for variable stiffness (VS) laminates designed by Direct Fiber Path Parameterization (DFPP) technique [76]. When the face panels are optimized as the variable stiffness panel, then, the optimization process would be better. The future aim is to further study this optimized composite panel as a variable stiffened panel.

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CURRICULUM VITAE

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Education:

2020-Continuing **Master of Engineering**

Ankara Yıldırım Beyazıt University /Ankara-Turkey

Department of Metallurgical and Materials Engineering

-I have chosen material science to excellence my material knowledge since all of the great engineering applications depend on material knowledge, especially, aerospace industries' applications. As a result of my experience and knowledge from B.Eng. Education, my related thesis will dwell on FEA-dependent optimization of honeycomb cored sandwich composite material which is used for increasing moment of inertia to prevent buckling in thin-walled structures such as aircraft wings.

2013-2019 **Bachelor of Engineering – Major**

Middle East Technical University /Ankara-Turkey

Department of Aerospace Engineering

-Most of my knowledge gained from the department is about fluid flow, structural mechanics and their computational analyses. During my education, I have aimed at computational sciences as my working area. I have chosen lots of elective courses according to this aim. I have gained good knowledge about both CFD & FEA.

Work Experience:

10.2021- Still working **Aeromechanical Analysis Engineer**

TR-Motor Propulsion Systems Industries Inc. Universiteler district, 1596th avenue, Hacettepe Technopark, F block, Floor 1-2-3 Çankaya / Ankara / Turkey

-I have been working in the structural analysis and integrity team (SAI) for classified APU & ATS projects. I have been completing static, modal, fatigue, and fracture mechanics analyses with the Ansys Workbench software for turbine and compressor blades. With the blade analysis results, I give feedback to aerodynamics and mechanical design teams on how to change and optimize the design in order to provide stress reduction on blades and stress equalization across the blade and disc on the critical points. My feedbacks depend on the untwist torque and bending stress reduction on the blades. In order to provide this, I give feedback for lean and tilt angles and hub-to-tip blade mean thicknesses. Furthermore, I have been completing disc burst analyses (radial burst & hoop burst) with FEA and hand calculations for disc burst rotational speed margins. To complete them, I have been doing research for reliable methods and imply them into the knowledge and methodology library of the company as best practices and in-house tools developed by Excel VBA(I have developed blade stress and 2D conservative crack stress intensity tools). None of the information about the project is free to share.

07.2021- 10.2021 **Structural Analysis Engineer**

Turkish Aerospace Industries Inc., Fethiye Aerospace Avenue No:17, 06980 Kahramankazan / Ankara / Turkey

-I have worked for a classified 5th generation fighter jet project as an external consultant staff by IDAK AERO (an International consulting company for defense and aerospace industries). I have completed airframe fastener sizing and evaluation under critical load cases with Hypermesh, Nastran, Hypersizer and self-developed tools. None of the information about the project is free to share.

08.2020-06.2021 **Analysis & Simulation Engineer**

KIRPART Automotive Parts Industry, Gedelek Town Gedelek Street. No:470, 16800 Orhangazi / Bursa / Turkey

-I have worked on both finite element analysis and computational fluid dynamics analysis with various ANSYS workbench software modules. I have completed modal, static and harmonic vibration analyses for internal combustion engine thermostats as FEA. Also, I have designed turbomachinery centrifugal pump impellers with the help of ANSYS VISTA (CPD) & BLADE-GEN and analyzed those pumps with ANSYS CFX according to expectations in terms of effective pressure rise, hydraulic efficiency and cavitation. Also, I have completed internal combustion engine thermostats' head loss CFD analyses with ANSYS FLUENT and gave feedback to the designers. In PORSCHE's new 9A3 engine project, I have designed the engine's water pump with the instructions and guidance of Prof. Dr. Frank Hendrik Wurm (The turbomachinery flow consultant of the PORSCHE A.G. and head of the University of Rostock in Germany) and this project provided me very good knowledge about turbomachinery flow.

07.2019-01.2020 **Computer-Aided Engineering Analyst**

EnginSoft Turkey, 2224th Street. No:1 ANKARA TEKNOPARK, 06378 İvedik OSB / Yenimahalle / Ankara / Turkey

-I have worked on finite element analysis with the Ansys FEA program on the different structure types those are turret, supporter structures carrying structures, etc. I have done static, modal, random vibration, transient (shock), fatigue, damage tolerance analyses on the structures having different elasticity types which are isotropic, hyperelastic and orthotropic Also, I have completed topology and parametric optimizations.(composite material).

02.2019-07.2019 **Candidate Engineer**

EnginSoft Turkey, 2224th Street. No:1 ANKARA TEKNOPARK, 06378 İvedik OSB / Yenimahalle / Ankara / Turkey

-I have worked on finite element analyses by ANSYS Workbench software on the physics of static-structural, modal, random vibration, harmonic vibration analysis and fatigue analysis.

06.2018-07.2018 **Internship**

MS-Spektral mechanical design team in Hacettepe University Technopark

- Worked on mechanical design with CATIA V5-R19. I have been a part of having ISO quality certificates.

06.2016-07.2016 **Internship**

SunExpress Line Maintenance Planning Department in Antalya Airport

-Worked on checks of the line maintenance packages, FAA airworthiness directive with related ATA chapters.

Languages:

Turkish: Native

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