بسم الله الرحمن الرحيم

Sudan University of Science and Technology

Faculty of Engineering

Aeronautical Engineering Department

Manufacturing Technology of Composite Material Structure

Thesis Submitted in Partial Fulfillment of the Requirements for the Degree of Bachelor of Engineering. (BEng Honor)

By:

1. ELSADIG HAIDER BASHRA MUSTAFA
2. MANSOR IBRAHIM MANSOR HAMID
3. MARWA FATHI ELTAHER IBRAHEEM
4. OMER AHMED MOHAMMED OMER

Supervised By:

GENERAL ENG. ABD ALRAHIM SAAD OMER

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الآيه

"أَمْ حَسِبْتُمْ أَن تَدْخُلُوا الْجَنَّةَ وَلَمَّا يَأْتِكُم مَّثَلُ الَّذِينَ خَلَوْا مِن قَبْلِكُم ۖ مَّسَّتْهُمُ الْبَأْسَاءُ وَالضَّرَّاءُ وَزُلْزِلُوا حَتَّىٰ يَقُولَ الرَّسُولُ وَالَّذِينَ آمَنُوا مَعَهُ مَتَىٰ نَصْرُ اللَّهِ ۗ أَلَا إِنَّ نَصْرَ اللَّهِ قَرِيبٌ"

البقرة - الآية 214

ABSTRACT

This project aims to manufacture an aileron for **SAFAT 01** by using composite material (**carbon/epoxy**) and analyze the structure in order to reduce the high weight due to using other materials.

This model sketched by using (**CATIA**) software program then exported into (**PATRAN**) software program which is used as a model to build the finite element model.

The model has been divided into five groups; upper spar-flange, lower spar-flange, spar-web, skin, rib-web. Each rib was divided into two part front and rear.

Then the parts were loaded, after that the model was submitted to (**NASTRAN**) software program. However the result has shown that all the structure (skin, spar and rib-web) led to over design condition, so we re-analyzed without spar. The results of re-analyzed shows that the structure of skin plus rib-web only is stronger enough to carry the load applied to the aileron structure, so that we remove the spar to avoid the over design condition, found that the resultant deformation values due to the applied load on the model is very small and it can be neglected and the using of this composite material (**carbon/epoxy**) in manufacturing this part is suitable.

Then a sample of this structure has been manufactured and the impact test, hardness test and positive material analyzing test has been applied to it.furthermore the result of the entire tests exposed that the sample of this structure is valid to use in the aileron model.

التجريد

يهدف هذا المشروع ألي تصنيع الجنيح لطائرة صافات (01) بأستخدام المواد المركبة (**Carbon/epoxy**) وتحلل البنية لتقليل الوزن العالي نتيجة لأستخدام مواد آخري.

صمم هذا النموذج بأستخدام برنامج الرسم (**CATIA**) وتم نقلة ألي برنامج (**PATRAN**) الذي يستخدم في النمذجة لبناء عناصر النموذج المحدد.

قسم هذا النموذج الي خمسة مجموعات ـ حافة العارضة العليا ، حافة العارضة السفلي ، العارضة ، الغطاء الخارجي ، الضلع. وكل ضلع قسم الي جزئين؛ جزء امامي وجزء خلفي.

ثم سلطت قوي علي الأجزاء. بعد ذلك اخضع النموذج لبرنامج التحليل (**NASTRAN**) و وجد ان الخمس مجموعات مجتمعه(العارضة ، الغطاء الخارجي و الضلع) ذات بنيه عاليه جدا اكثر من القوه المسلطه مما ادي الي حاله التصميم الزائد، لذلك تمت اعاده التحليل مع ازاله العارضه. اوضحت نتائج التحليل ان بنيه الغطاء الخارجي والضلع فقط قويه كفايه لتتحمل القوى المسلطه علي بنيه الجنيح، لذلك تم ازاله العارضه من التصميم لتجنب حاله التصميم الزائد. وجد ان قيم نتائج التشوه الناتجه عن القوى المسلطه علي بنيه النموذج صغيره جدا و يمكن تجاهلها و استخدام المواد المركبه (**carbon/epoxy**) في عمليه تصنيع هذا الجزء (الجنيح) مناسبة.

ثم تم تصنيع عينه من البنية المعنية واجريت عليها اختبار امتصاص الصدمه، اختبار الصلاده و اختبار تحديد العناصر المكونه للماده وتم التاكد من نتائج الإختبارات بأن هذه البنية مناسبة لإستخدامها في النموذج.

Acknowledgement

It is a pleasure to thank those who were supporting us during this project...

First, to ALLAH who created us….
To our families whom built us to face this life …
And to my great teacher Prof. ABDERAHEEM SAAD and engineering.
ABDELMAGID ADRES. I would thank you for your patience and support.
Greater thanks to MR. AHMED MOKHTAR to help us.

Dedication

I would like to dedicate my research to all those who helped me through this enlightening experience, and to my teachers, father, mother and best friend, and my brother all within the span of this research, who have made countless sacrifices in the interest of this research and in the interest of my success.

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Symbol	**Definition**
B	Beam curvature at a given
E_{11}	Young's moduli in the 1 and 2 directions, respectively
E_{22}	Young's moduli in transverse direction
Ep	Ply thickness
e	Overall thickness of laminate
F_{11}	Strength in fiber direction
F_{22}	Strength in transverse direction
F_b	Buckling allowable
$F_{mat,s}$	Material shear allowable for ultimate
$F_{mat,t}$	Material tensile allowable for ultimate
G_{12}	Shear modulus
h	Height of spar
K	Fiber coordinate system
$\kappa_x, \kappa_y and \kappa_s$	The Kirchhoff shear forces
$K_w K_{NO,w}$	The factor adesign varible (at each station)
K_s	IS skin ratio
l_r	Rib spacing
ll_{aero}	Toatal aerodynamic load
M_x	Bending moment tensor
N_x	Flux tensor
P	Is the load on the cover panel due to the bending moment.
Q	Stiffness matrix
R	Beam radius of curvature at a given point
S_{ALT}	Shear load at the station
SF	Load factor
T_{LRA}	Torsion load at the station
t_e	Skin thickness

t_w	Thickness spar web
$t_{rib,w}$	Thickness rib web
$t_{rib,c}$	Thickness rib cap
E_f, G_f, E_f, ν_f	Properties of fiber
E_m, G_m, E_m, ν_m	Properties of matrix material
t_S	Skin thickness
ν_{12}	Poisons ratio
σ_1	Longitudinal strength (both tensile and compressive).
σ_2	Transverse strength (both tensile and compressive).
τ_{12}	Shear strength
$\varepsilon_x, \varepsilon_y, \gamma_{xy}$	Strains at any point

1 Chapter One: Introduction

1.1 Overview

A composite material can be defined as a combination of two or more materials that results in better properties than when the individual components are used alone (e.g. carbon fiber ceramic composite CFCC, glass fiber reinforcement plastic GFRP, carbon fiber reinforcement plastic CFRP, aramid fiber reinforcement plastic AFRP). The whole purpose of manufacturing is to manufacture aircrafts structures using composites materials by used the method of manufacturing which it divided to manual lay-up, spray-up, filament winding, pultrusion and resin transfer molding. The analysis structures of composite ensure that the composite is better than metals to use in aileron structure (carbon/epoxy).

1.2 Problem Statement

The material (aluminium alloy) which used in made of aileron structure made it high weight.

1.3 Proposed Solution

Using composite material in made the aileron structure (Carbon /epoxy).

1.4 Aim and Objectives

1.4.1 Aim

Improvement of industry production, using best method of manufacturing.

1.4.2 Objectives

1. Manufacturing of SAFAT (01) aileron using (Carbon/epoxy) and analysing structure.
2. Comparative between the material properties of aluminium alloy and (Carbon /epoxy) to use in made aileron.
3. Evaluate the performance of the composite material structure in different industrial fields.

1.5 Methodology

All about composite materials have been studied and compared with metals, then the aileron has been selected as a model for manufacturing. All data of an aileron for a specific aircraft has been collected after that the drawing of the model using CATIA V5R18 software has been done and then it’s exported to PATRAN 2010.1 software, then the structure analysis for the model using NASTRAN software has been done. However, the model being able to manufacturing. The results have been obtained from analysis were compared with the metals to ensure the composite profitableness. The results were discussed as well as recommendations were presented.

1.6 Thesis outlines

Chapter One: Introduction

Chapter Two: Literature Review

Chapter Three: Manufacturing Process

Chapter Four: Calculation

Chapter Five: Results and Discussion

Chapter Six: Conclusion and Recommendations

2 Chapter Two: Literature Review

2.1 History and Background

A composite material can be defined as a combination of two or more materials that results in better properties than the when the individual components are used alone. As opposed to metal alloys, each material retains its separate chemical, physical and mechanical properties. The two constituents are normally a fiber and a matrix. Typical fibers include glass, aramid and carbon, which may be continuous or discontinuous. Matrices can be polymers, metals or ceramics.[1]

Since the early-1920s, airframes have been built largely out of metal, aluminum has been the material of choice. When high performance composites (i.e., first boron and then carbon fibers) started being developed in the mid-1960s and early-1970s, the situation started changing. The earliest developers, and users, of composites were the military. The earliest production usage of high performance composites was on the empennages of the F-14 and F-15 fighter aircraft. Boron/epoxy was used for the horizontal stabilators on both aircraft, and for the rudders and vertical fins on the F-15. In the mid-1970s, with the maturity of carbon fibers, a carbon/epoxy speed brake was implemented on the F-15. While these early applications resulted in significant weight savings 20%, they accounted for only small amounts of the airframe structural weight.[2]

However, as shown in Figure 2.1, composite usage quickly expanded from only 2% of the airframe on the F-15 to as much as 27% on the AV-8B Harrier by the early-1980s. Significant applications included the wing (skins and substructure), the forward fuselage, and the horizontal stabilator, all fabricated from carbon/epoxy. While the number of composites used on the AV-8B was somewhat on the high side, most modern fighter aircraft contain over 20% composite structure. Typical weight savings usually range from 15 to 25%, depending on the particular piece of structure, with about 20% being a good rule of thumb. Similar trends have been followed for commercial aircraft, although at a slower and more cautious pace. Until recently, Airbus has been somewhat more aggressive in using composites than Boeing, primarily for00 horizontal stabilizers and vertical fins on their A300 series of aircraft. However, Boeing recently made a major commitment to composites, when it decided to use upwards of 50% on its new 787, which includes both a composite wing and fuselage, as depicted in Figure 2.2.[2]

Figure 2.1: composites evolution in fighter aircraft.[3]

Source: The Boeing Company

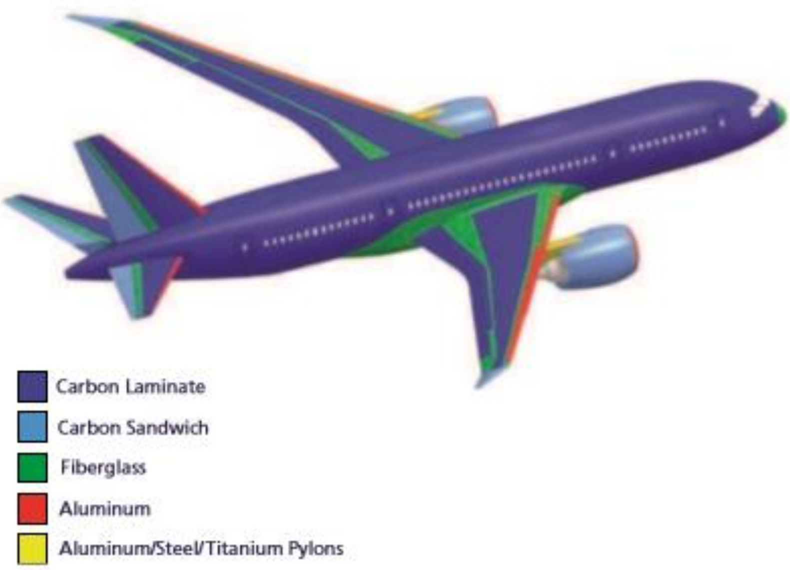


Figure 2.2: Boeing 787 material distribution.

2.2 Composite Materials

A composite is a structural material that consists of two or more combined constituents that are combined at a macroscopic level and are not soluble in each other. One constituent is called the reinforcing phase and the one in which it is embedded is called the matrix. The reinforcing phase material may be in the form of fibers, particles, or flakes. The matrix phase materials are generally continuous.[4]

2.3 Advanced composite materials

Advanced composites are composite materials that are traditionally used in the aerospace industries. These composites have high performance reinforcements of a small diameter in a matrix material such as epoxy and aluminum.[4]

The properties that can be improved by forming a composite material are strength, stiffness, corrosion resistance, wear resistance, attractiveness, weight, fatigue life, temperature dependent behavior, thermal insulation, thermal conductivity and acoustical insulation.[4]

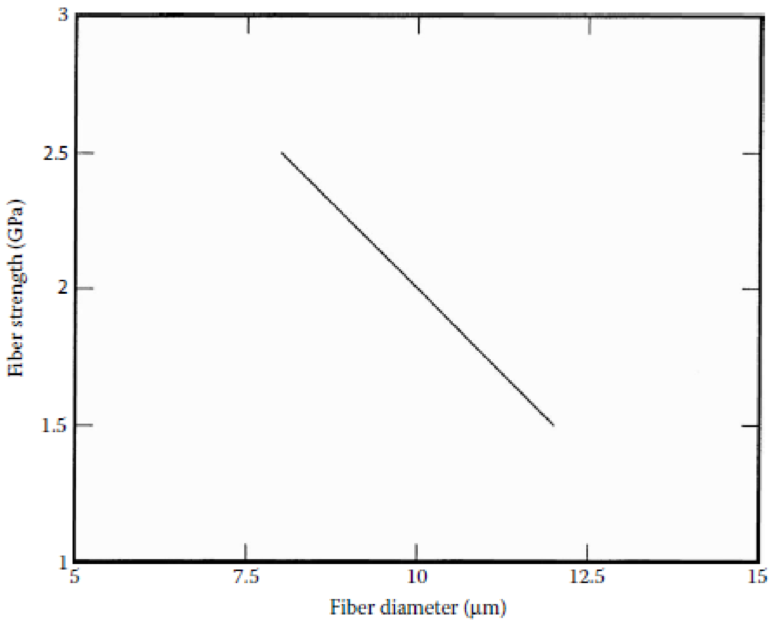


Figure 2.3: fiber strength as a function of fiber diameter for carbon fibers.[4]

2.4 Composite Applications in Aircraft Structures

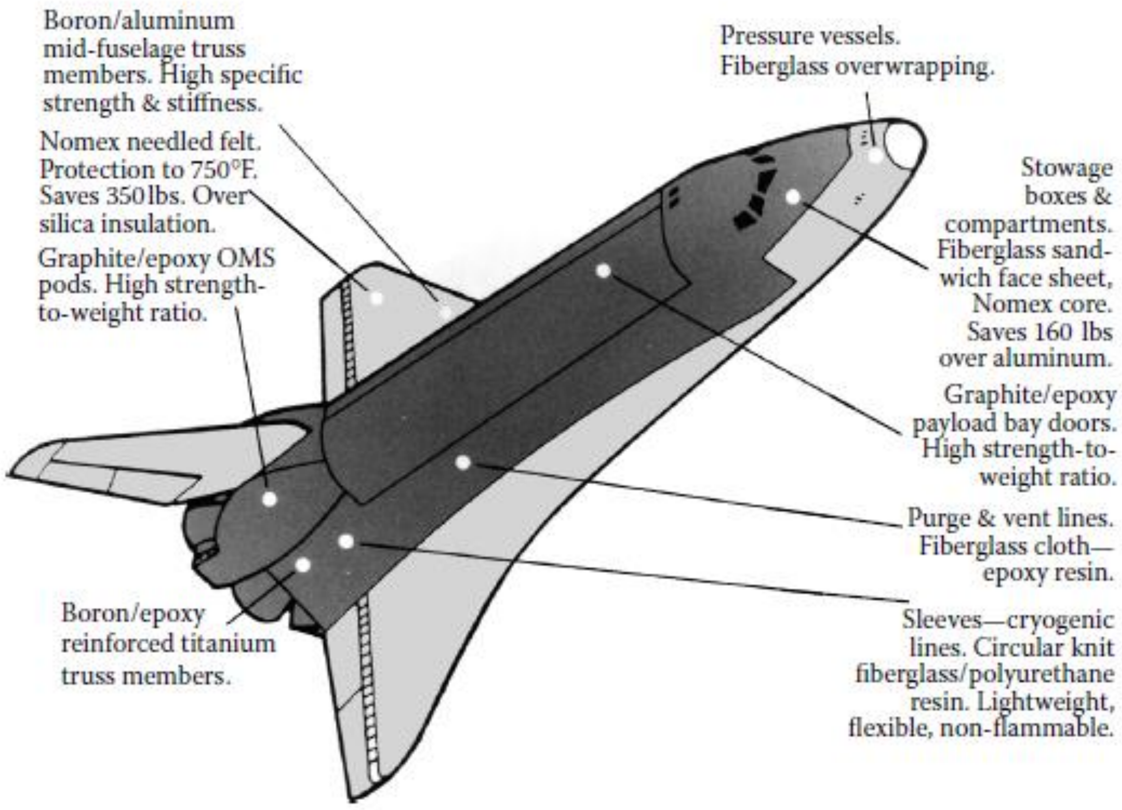


Figure 2.4: use of composites in the space shuttle.

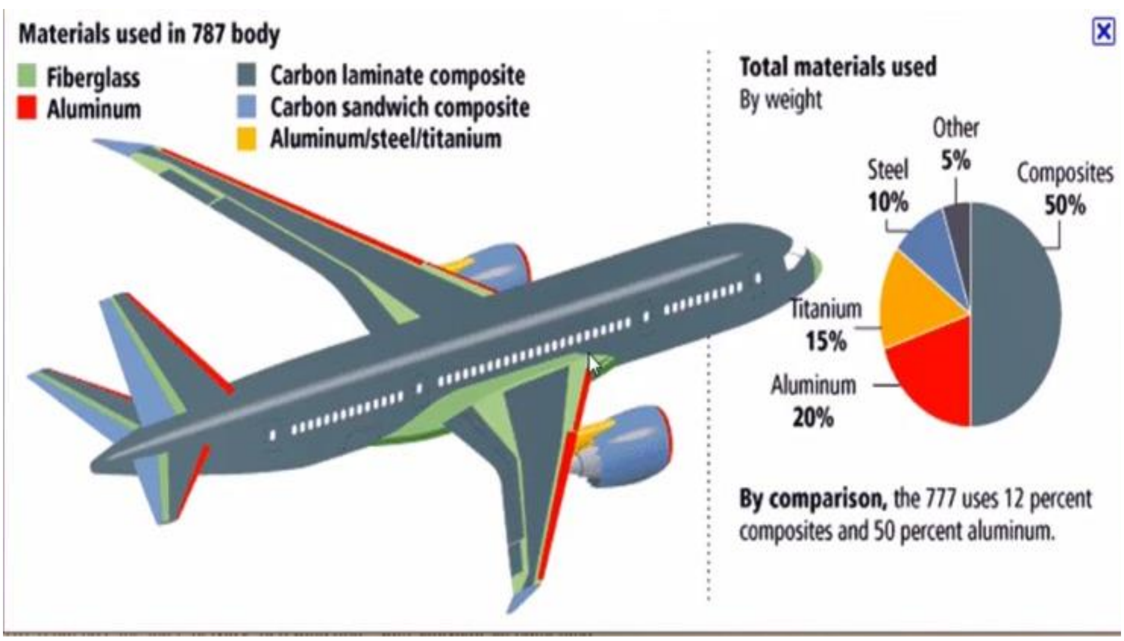


Figure 2.5: use of composites in the dream linear B787.

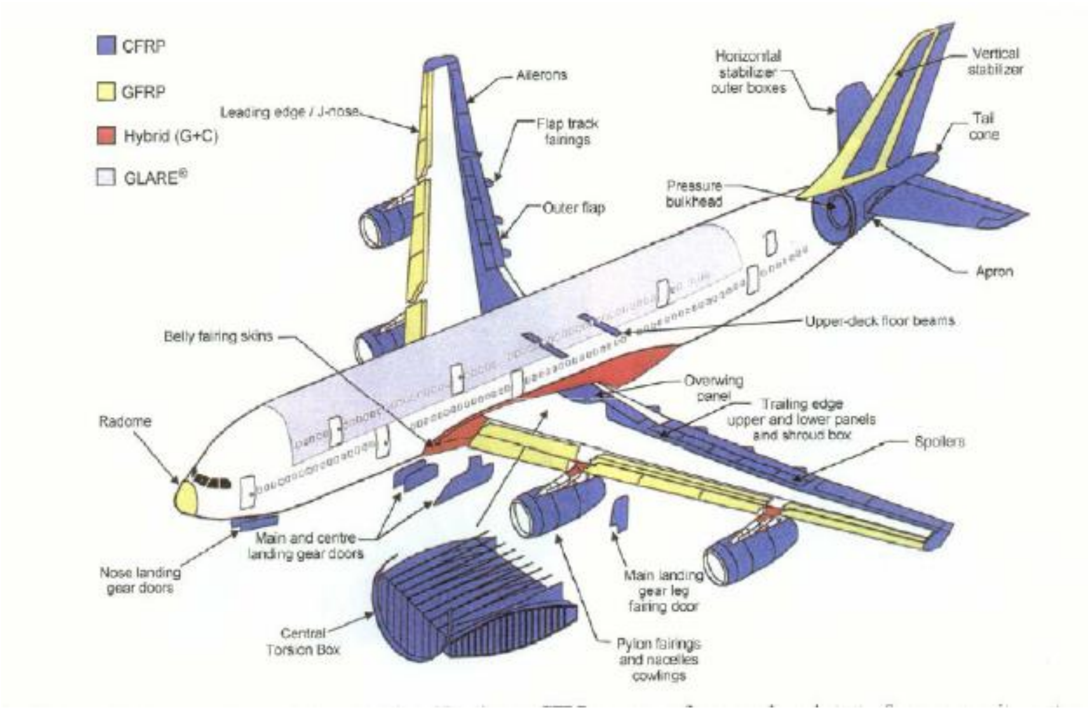


Figure 2.6: use of composite in A380.[5]

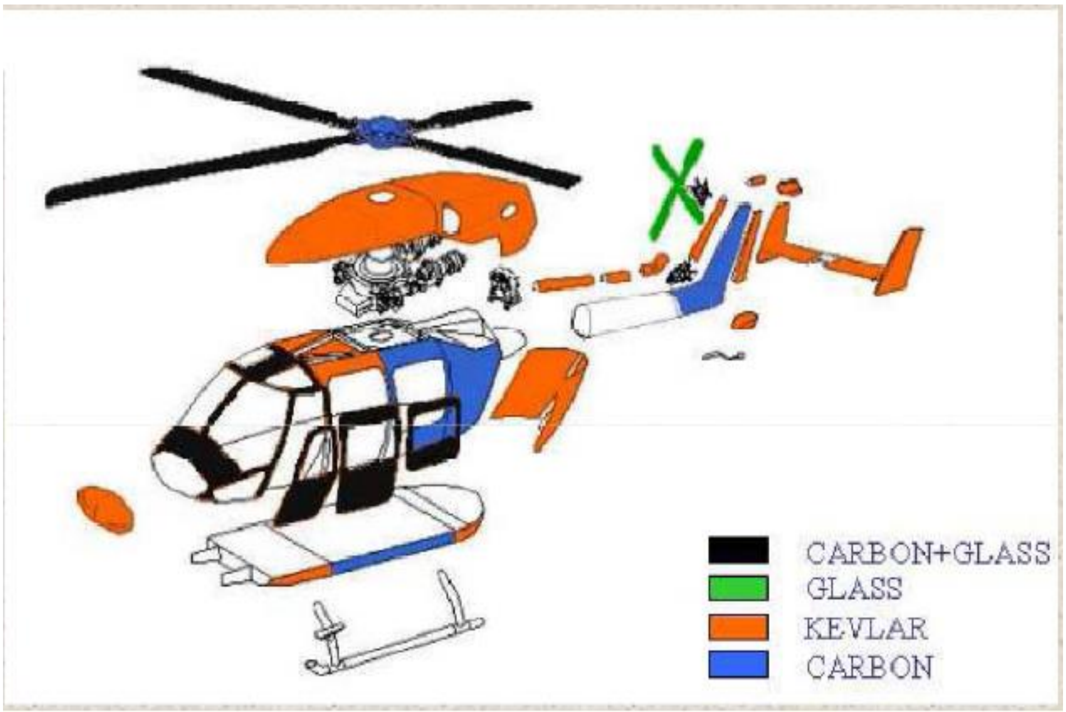


Figure 2.7: Use of composites in LCH

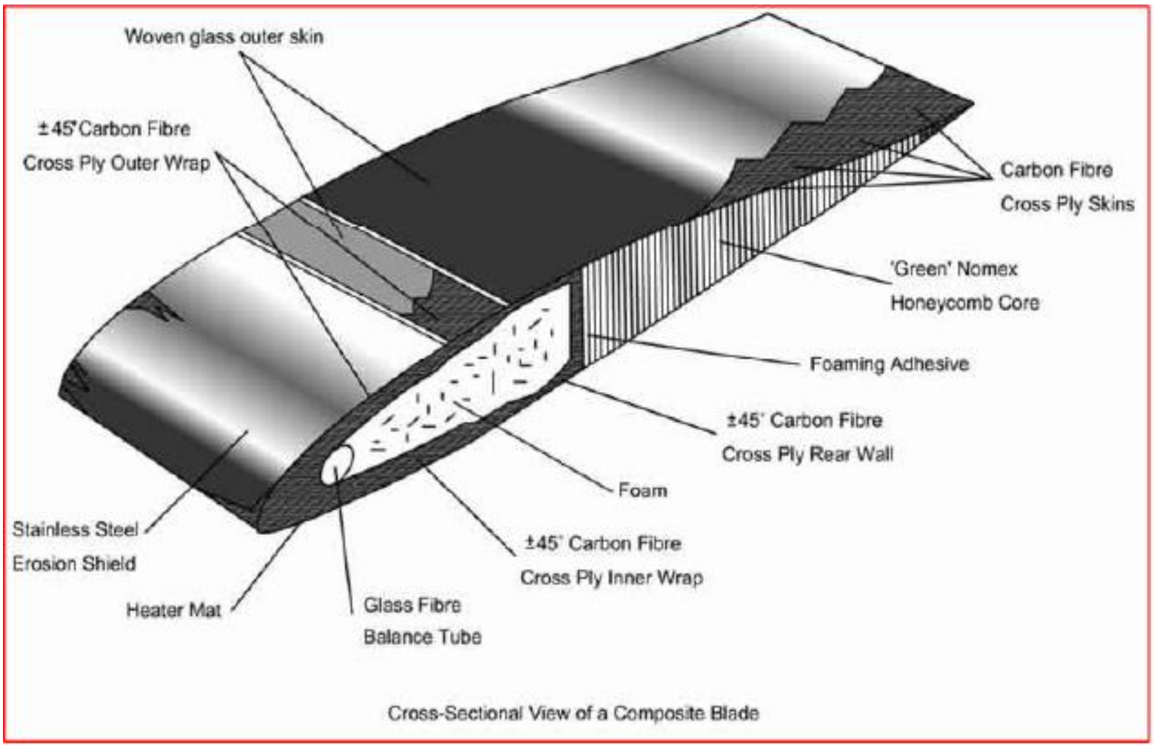


Figure 2.8: Helicopter Blade

2.5 The Advantages and Disadvantages of Composite Materials[5]

2.5.1 Advantages of Composite Materials

1. Light weight.
2. Highly resistance to corrosion.
3. High resistance to fatigue damage.
4. High strength
5. High stiffness
6. Exceptional formability
7. Electrical insulation
8. Thermal insulation
9. Acoustical insulation

2.5.2 Disadvantages of Composite Materials

1. High cost of raw materials and fabrication.
2. Composites are brittle and thus are more easily damageable.
3. Transverse properties may be weak.
4. Matrix is weak, therefore, low toughness.
5. Reuse and disposal may be difficult.
6. Health hazards during manufacturing, during and after use.

7. Joining to parts is difficult.
8. Repair introduces new problems, for the following reasons:
    a) Materials require refrigerated transport and storage and have limited shelf life.
    b) Hot curing is necessary in many cases requiring special tooling.
    c) Curing takes time.
9. Analysis is difficult.
10. Matrix is subject to environmental degradation

2.6 Composites Versus Metals Comparison[5]

Table 2.1: comparative between composites and metals.

Condition	**Comparative behavior relative to metals**
Load-strain relationship	More linear strain to failure
Notch sensitivity	Greater sensitivity
Static	Less sensitivity
Fatigue	Weaker
Transverse properties	Higher
Mechanical properties variability	Higher
Fatigue strength	Greater
Sensitivity to corrosion	Much less
Damage growth mechanism	In-plane delamination instead of through thickness cracks

2.7 Classification of Composite Materials

Generally, the constituents of composite are the reinforcements and the matrices.

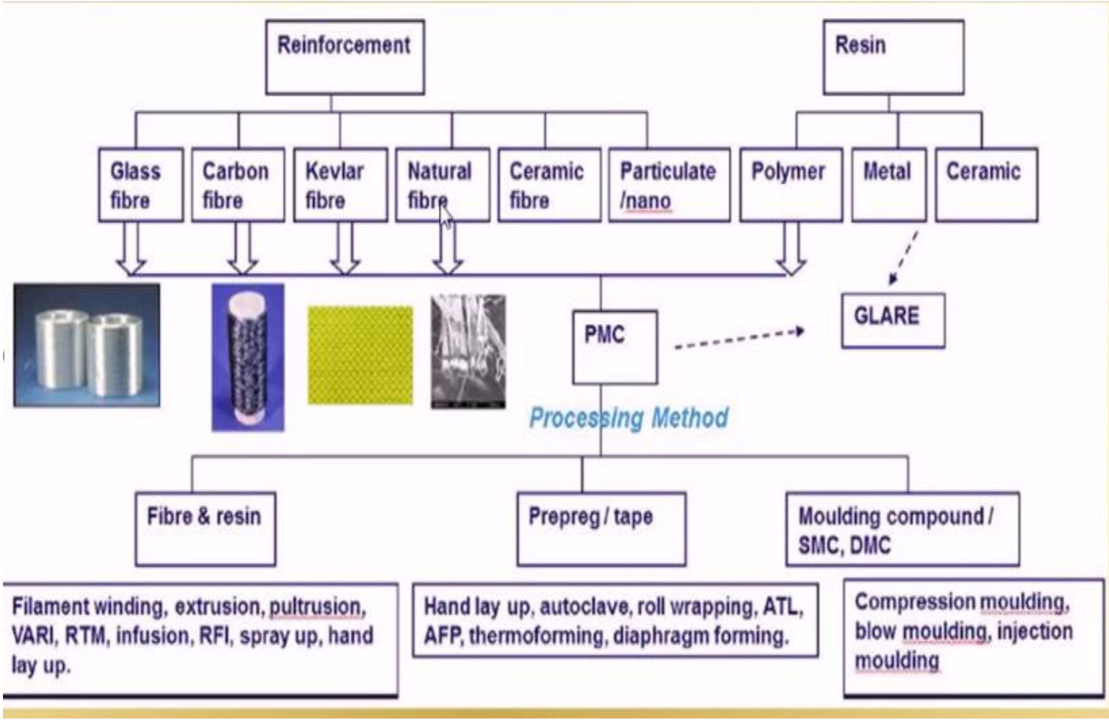


Figure 2.9: classification of composite material

2.8 Matrices

The continuous phase is the matrix, which is a polymer, metal, or ceramic. Polymers have low strength and stiffness, metals have intermediate strength and stiffness but high ductility, and ceramics have high strength and stiffness but are brittle.[4]

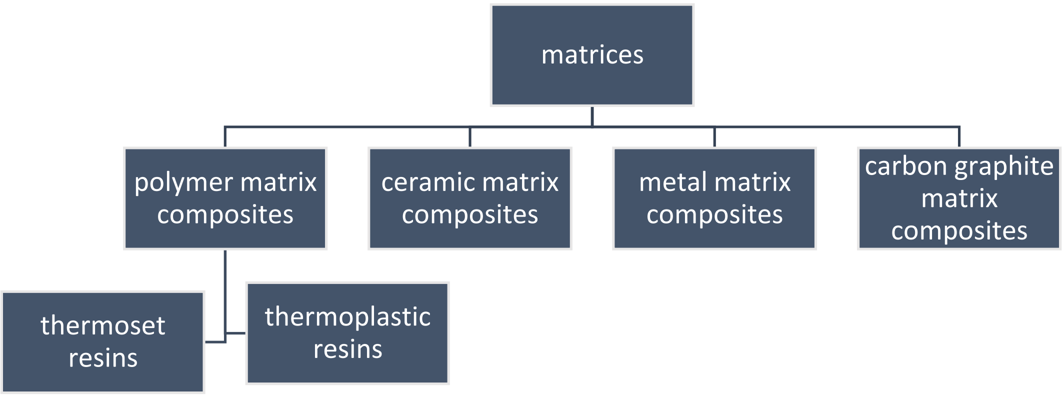


Figure 2.10: classification of matrix

The matrix (continuous phase) performs several critical functions, including maintaining the fibers in the proper orientation and spacing and protecting them from abrasion and the environment. In polymer and metal matrix composites that form a strong bond between the fiber and the matrix, the matrix transmits loads from the matrix to the fibers through shear loading at the interface. In ceramic matrix composites, the objective is often to increase the toughness rather than the strength and stiffness; therefore, a low interfacial strength bond is desirable.[4]

2.8.1 Polymer Matrix Composite

Matrices for polymeric composites can be either thermosets or thermoplastics. Thermoset resins usually consist of a resin (e.g., epoxy) and a compatible curing agent. When the two are initially mixed they form a low-viscosity liquid that cures because of either internally generated (exothermic) or externally applied heat. The curing reaction forms a series of cross-links between the molecular chains so that one large molecular network is formed, resulting in an intractable solid that cannot be reprocessed on reheating. On the other hand, thermoplastics start as fully reacted high-viscosity materials that do not cross-link on heating. On heating to a high enough temperature, they either soften or melt, so they can be reprocessed several times.[4]

2.8.1.1 Thermoset Resins

a) Epoxy

High performance matrix systems for primarily continuous fiber composites. Can be used at temperatures up to 250-275 F. better high temperature performance than polyesters and vinyl esters.[4]

b) Vinyl esters

Like polyesters but are tougher and have better moisture resistance.

c) Polyesters

Used extensively in commercial applications. Relatively inexpensive with processing flexibility. Used for continuous and discontinuous composites.

d) Bismaleimdes

High temperature resin matrices for use in the temperature range of 275-350F with epoxy-like processing. Requires elevated temperature post cure.

e) Polyimides

Very high temperature resin system for use at 550-600F. Very difficult to process.

f) Phenolics

High temperature resin systems with good smoke and fire resistance. Used extensively for aircraft interiors. Can be difficult to process.

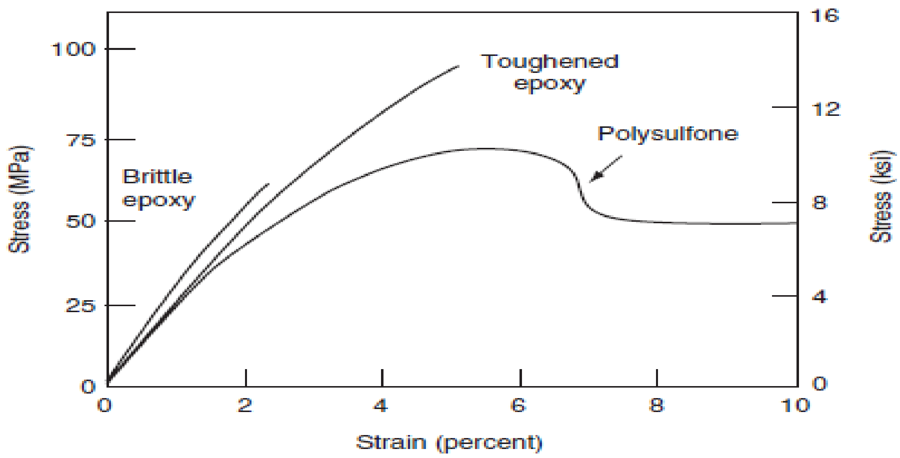


Figure 2.11:Tensile stress–strain diagrams of a thermoset polymer (epoxy) and a thermoplastic polymer (polysulfone).[6]

A) Epoxy Resins

Epoxies are the most common matrix material for high-performance composites and adhesives. They have an excellent combination of strength, adhesion, low shrinkage and processing versatility. Commercial epoxy matrices and adhesives can be as simple as one epoxy and one curing agent; however, most contain a major epoxy, one to three minor epoxies and one or two curing agents. The minor epoxies are added to provide viscosity control, improve high-temperature properties, lower moisture absorption or improve toughness. There are two major epoxies used in the aerospace industry[6]:

I. Di-glycidyl ether of Bisphenol A (DGEBA), which is used extensively in filament winding, pultrusion and some adhesives.

II. Tetra-glycidyl methylene di-aniline (TGMDA), also known as tetraglycidyl-4,4"-diaminodiphenylmethane (TGGDM), which is the major epoxy used for a large number of the commercial composite matrix systems.

Epoxy matrix, has the following advantages over other thermoset matrices[6]

a. Wide variety of properties, since many starting materials curing agents, and modifiers are available
b. Absence of volatile matters during cure
c. Low shrinkage during cure
d. Excellent resistance to chemicals and solvents
e. Excellent adhesion to a wide variety of filler s, fibers, and other substrates

2.8.1.2 Thermoplastic Resins

Before considering the potential advantages of thermoplastic composite materials, it is necessary to understand the difference between a thermoset and thermoplastic. As shown in figure 2.10, a thermoset cross-link during cure to form a rigid intractable solid. Prior to cure the resin is a relatively low molecular-weight semi-solid that melts and flows during the initial part of the cure process. As the molecular weight builds during cure, the viscosity increases until the resin gels and then strong covalent bond cross-links form during cure. Due to the high cross-link densities obtained for high performance thermoset systems, they are inherently brittle unless steps are taken to enhance toughness.[7]

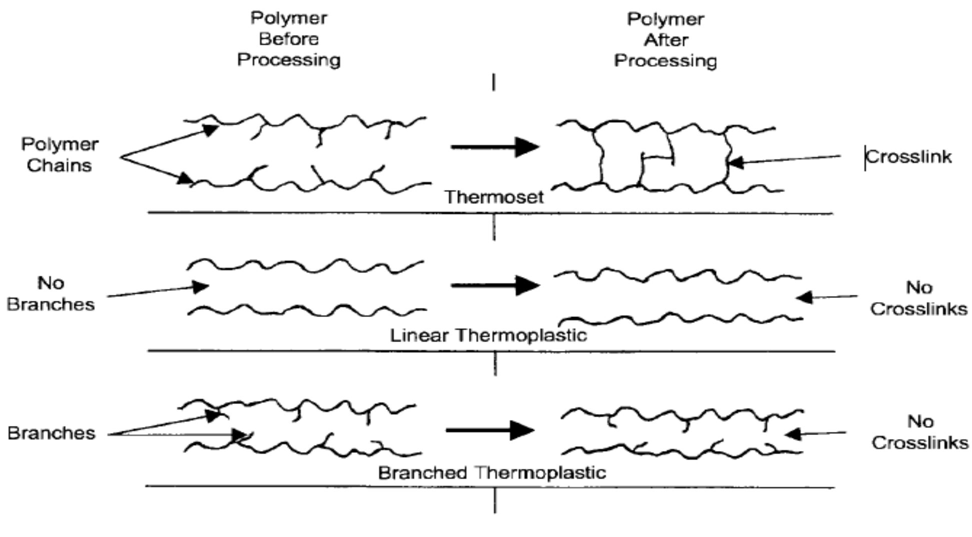


Figure 2.12: comparison between thermoset and thermoplastic polymer structure.

On the other hand, thermoplastics are high molecular-weight resins that are fully reacted prior to processing. They melt and flow during processing but do not form cross-linking reactions. Their main chains are held together by relatively weak secondary bonds.[3]

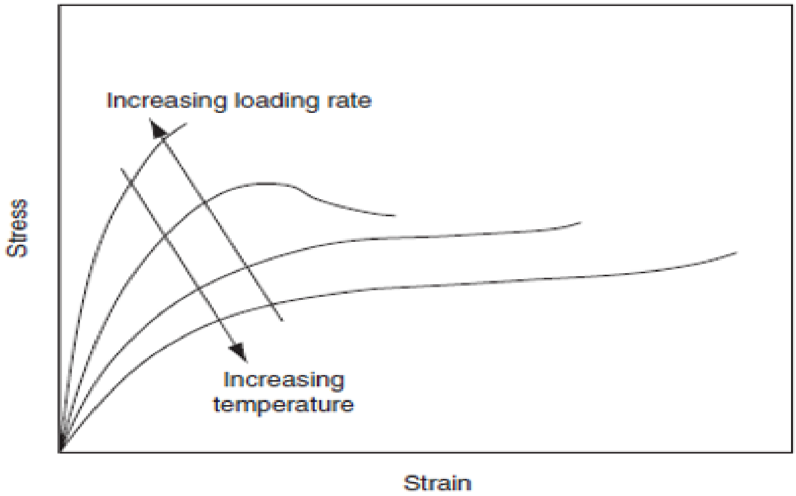


Figure 2.13: Effect of loading rate and temperature on the stress-strain behavior of polymeric solids.

[Figure 2.12]. Shows the effects of temperature and loading rate on the stress–strain behavior of polymeric solids. At low temperature s, the stress–strain behavior is much like that of a brittle material. The polymer may not exhibit any signs of yielding and the strain-to-failure is low. As the temperature is increased, yielding may occur; but the yield strength decreases with increasing temperature.

The strain-to-failure, on the other hand, increases with increasing temperature, transforming the polymer from a brittle material at low temperatures to a ductile material at elevated temperatures.

The effect of loading rate on the stress–strain behavior is opposite to that due to temperature [Figure 2.12]. At low loading rates or long durations of loading, the polymer may behave in a ductile manner and show high tough ness. At high loading rates or short durations of loading, the same polymer behaves in a rigid, brittle (glass-like) manner.[8]

The most important advantage of thermoplastic polymers over thermoset polymers is their high impact strength and fracture resistance, which in turn impart an excellent damage tolerance characteristic to the composite material. In general, thermoplastic polymers have higher strain-to-failure than thermoset polymers, which may provide a better resistance to matrix micro cracking in the composite laminate. Other advantages of thermoplastic polymers are[6]

1. Unlimited storage (shelf) life at room temperature
2. Shorter fabrication time
3. Post formability (e.g., by thermoforming)
4. Ease of joining and repair by welding, solvent bonding, and so on
5. Ease of handling (no tackiness)
6. Can be reprocessed and recycled

Table 2.2: Maximum Service Temperature for Selected Polymeric

Polymer	**Tg, 8C**	**Maximum Service Temperature, 8C (8F)**
Thermoset matrix		
DGEBA epoxy	180	125 (257)
TGDDM epoxy	240–260	190 (374)
Bismaleimides (BMI)	230–290	232 (450)
Acetylene-terminated polyimide (ACTP)	320	280 (536)
PMR-15	340	316 (600)
Thermoplastic matrix		
Polyether ether ketone (PEEK	143	250 (482)
Polyphenylene sulfide (PPS)	85	240 (464)
Polysulfone	185	160 (320)
Polyetherimide (PEI)	217	267 (512)
Polyamide-imide (PAI)	280	230 (446)
K-III polyimide	250	225 (437)

2.8.2 Metal Matrix Composite

Metal-matrix composites (MMCs), 4'7'8 with continuous or discontinuous fiber reinforcement have been under development for well over 30 years, but have yet to be widely exploited.[3]

The main MMCs based on continuous fibers, and their advantages and disadvantages compared with PMCs, are listed in Table 2.3. Potential aircraft applications of the MMCs include engine components, such as fan and compressor blades, shafts, and possibly discs, airframe components, such as spars and skins, and undercarriage components, such as tubes and struts.[3]

Table 2.3: Candidate Continuous Fiber MMCs Compared with PMCs

Promising systems	A. boron/aluminum alloy; silicon carbide/aluminum; alumina/aluminum B. silicon carbide/titanium; silicon carbide/titanium aluminide C. carbon/aluminum; carbon/magnesium (only for space applications)
Advantages	A. higher temperature capability, particularly titanium and titanium aluminide B. higher through-thickness strength, impact damage resistant C. higher compressive strength D. resistant to impact damage E. high electrical and thermal conductivity
Disadvantages	A. limited and costly fabrication technology B. difficult and inefficient joining technology C. limited in temperature capability by fiber/matrix chemical incompatibility D. prone to thermal fatigue: fiber/matrix expansion mismatch problem E. prone to corrosion, particularly with conducting fibers

2.8.3 Ceramic Matrix Composite

Ceramic-matrix composites (CMCs) summarized in [Table 2.4], offer the main long-term promise for high-temperature applications in gas turbine engines and for high-temperature airframe structures, although there are formidable problems to be overcome. The main requirement is for lightweight blades able to operate uncooled in environments around 1400°C.[3]

The main limitation is the unavailability of fibers with high-elastic moduli and strength, chemical stability, and oxidation resistance at elevated temperatures.[3]

For suitable reinforcement of ceramic matrices (such as alumina and silicon carbide or silicon nitride), the fiber must have high oxidation resistance at high temperature because micro cracking of the ceramic allows contact between the fibers and the external environment

Table 2.4: candidate matrix composites-advantages and disadvantages compared with PMCs.

Systems	a. silicon carbide/glass; silicon carbide silicon nitride b. carbon/carbon; carbon/glass c. alumina/glass
Advantages	a. high to very high temperature capability (500-1500 °C) b. resistant to moisture problems c. low conductivity d. low thermal expansion e. resistant to aggressive environments
Disadvantages	a. fabrication can be costly and difficult b. joining difficult c. relatively low toughness d. matrix microcracks at low strain levels

2.9 Types of Reinforcement Composite Materials

1. Fibrous composite material. (consist of fibers in a matrix)
2. Laminate composite material. (consist of layers of various materials)
3. Particulate composite material. (composed of particles in a matrix)
4. Combinations of some or all the first three types.

2.9.1 Fibrous Composite Materials

A fiber has a length that is much greater than its diameter. The length-to-diameter (l/d) ratio is known as the aspect ratio and can vary greatly.[4]

Continuous fibers have long aspect ratios, while discontinuous fibers have short aspect ratios.[4]

Continuous-fiber composites normally have a preferred orientation, while discontinuous fibers generally have a random orientation. Examples of continuous reinforcements [figure 2.14] include unidirectional, woven cloth, and helical winding, while examples of discontinuous reinforcements [figure 2.15] are chopped fibers and random mat [figure2.15].[4]

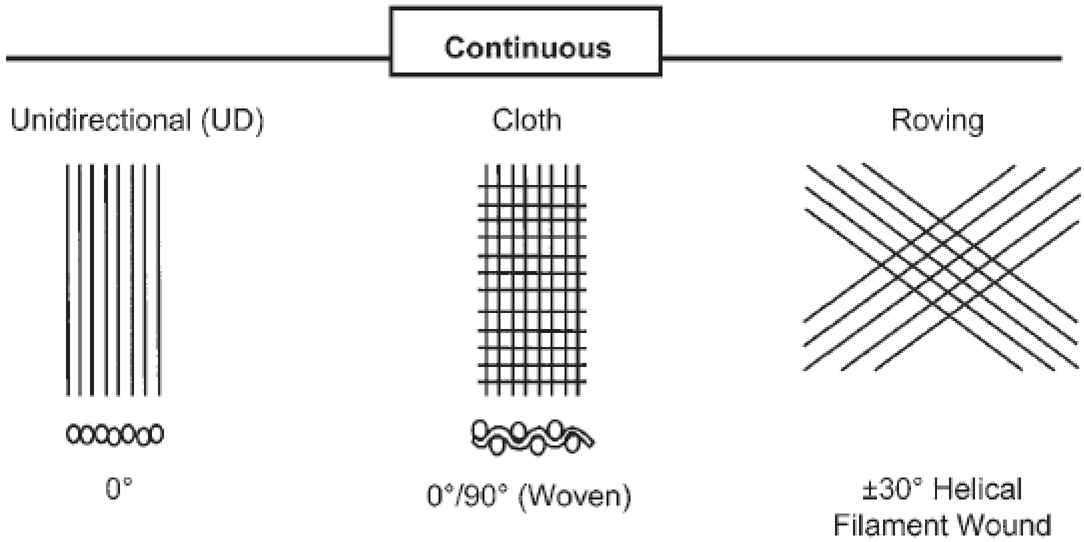


Figure 2.14: examples of continuous reinforcement.

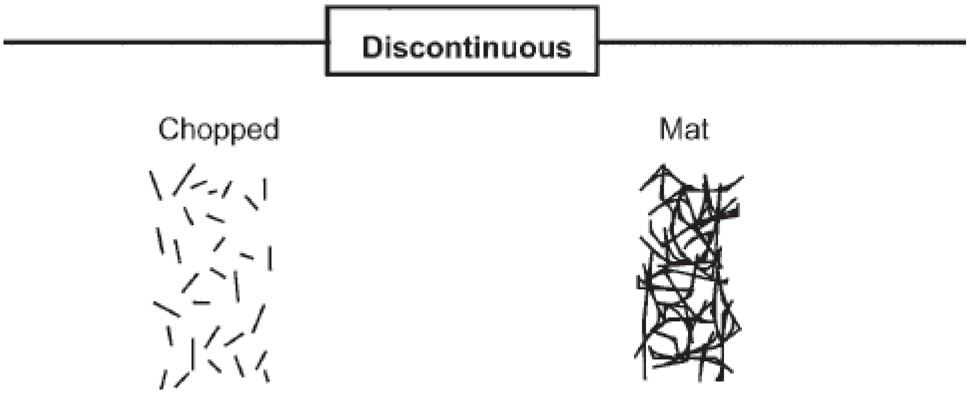


Figure 2.15: examples of discontinuous reinforcements

2.9.1.1 Typical fibers include

1. Glass fibers are the most widely used reinforcement due to their good balance of mechanical properties and low cost.[9]

2. **E-glass, or "electrical" glass**, is the most common glass fiber and is used extensively in commercial composite products. E-glass is a low-cost, high-density, low-modulus fiber that has good corrosion resistance and good handling characteristics.[9]
3. S-2 glass, or "structural" glass, was developed in response to the need for a higher-strength fiber for filament-wound pressure vessels and solid rocket motor casings. Its density value, performance level and cost lie between those of E-glass and carbon.[9]
4. Quartz fiber is used in many electrical applications due to its low dielectric constant; however, it is very expensive.[9]
5. Aramid fiber (e.g., Kevlar) is an extremely tough organic fiber with low density, and exhibits excellent damage tolerance. Although it has a high tensile strength, it performs poorly under compression. It is also sensitive to ultraviolet light and its use should be limited to long-term service at temperatures less than 350 °F.[9]
6. Carbon fiber contains the best combination of properties but is also more expensive than either glass or aramid. It has a low density, a low coefficient of thermal expansion (CTE) and is conductive. It is structurally very efficient and exhibits excellent fatigue resistance. It is also brittle (strain- to-failure less than

2 %) and exhibits low impact resistance. Being conductive, it causes galvanic corrosion if placed in direct contact with aluminum. Carbon fiber is available in a wide range of strength (300- 1000 ksi) and stiffness (modulus 30-145 msi). With this wide range of properties, carbon fibers are frequently classified as: high- strength, intermediate-modulus or high-modulus fibers.[9]

The terms carbon and graphite are often used to describe the same material. However, carbon fibers contain -95 % carbon and are carbonized at 1800-2700 ~ while graphite fibers contain -99% carbon and are first carbonized and then graphitized at temperatures between 3600 ~ and 5500 ~ In general, the graphitization process results in a fiber with a higher modulus. Carbon and graphite fibers are made from rayon, polyacrylonitrile (PAN) or petroleum-based pitch. PAN-based fibers produce the best combination of properties. Rayon was developed as a precursor prior to PAN but is rarely used today, due to its higher cost and lower yield. Petroleum-based pitch fibers were also developed as a lower cost alternative to PAN but are mainly used to produce high- and ultra- high-modulus graphite fibers. Both carbon and graphite fibers are produced as untwisted bundles called tows. Common tows sizes are l k, 3k, 6k, 12k and 24k, where k = 1000 fibers. Immediately after fabrication, carbon and graphite fibers are normally surface treated to improve their adhesion to the polymeric matrix. Sizing, often epoxies without a curing agent, are frequently applied as thin films (1% or less) to improve handle ability and protect the fibers during weaving or other handling operations.[9]

7. Boron fiber was the original high-performance fiber before carbon was developed. It is a large-diameter fiber that is made by pulling a fine tungsten wire through a long slender reactor where it is chemically vapor- deposited with boron. Since it is made one fiber at a time, rather than thousands of fibers at a time, it is very expensive. Due to its large diameter and high modulus, it exhibits outstanding compression properties. Among the disadvantages, it does not conform well to complicated shapes and is very difficult to machine.[9]

2.9.1.2 The following factors should be considered when choosing between glass, aramid and carbon fibers[9]

1. Tensile strength: If tensile strength is the primary design parameter, E-glass may be the best selection because of its low cost.[9]

2. Tensile modulus: When designing for tensile modulus, carbon has a distinct advantage over both glass and aramid.[9]
3. Compression strength: If compression strength is the primary requirement, carbon has a distinct advantage over glass and aramid. Due to its poor compression strength, aramid should be avoided.[9]
4. Compression modulus: Carbon fibers are the best choice, with E- glass having the least desirable properties.[9]
5. Density: Aramid fibers have the lowest density, followed by carbon and then S-2 and E-glass.[9]
6. CTE: Aramid and carbon fibers have a CTE that is slightly negative, while S-2 and E-glass are positive.[9]
7. Impact strength: Aramid fibers have excellent impact resistance, while carbon is brittle and should be avoided. It should be noted that the matrix also has a significant influence on impact strength.[9]
8. Environmental resistance: Matrix selection has the biggest impact on composite environmental resistance. However, aramid fibers are degraded by ultraviolet light and the long-term service temperature should be kept below 350 °, carbon fibers are subject to oxidation at temperatures exceeding 700 ° although long-term 1000 h thermal oxidation stability tests in polyimides have shown strength decreases in the 500-600 ° range; and glass sizing tend to be hydrophilic and absorb moisture.[9]
9. Cost: E-glass is the least expensive fiber, while carbon is the most expensive. The smaller the tow size, the more expensive the carbon fiber. Larger tow sizes help reduce labor costs because more material is deposited with each ply. However, large tow sizes in woven cloth can increase the chances of voids and matrix micro-cracking due to larger resin pockets.[9]

2.9.2 Structural Composite Materials

Composite materials structure forms are

1. Laminated composite structure
2. Sandwich structure

1. Laminate Composite Material

Continuous-fiber composites are laminated materials [Figure 2.16] in which the individual layers, plies or laminate are oriented in directions that enhance the strength in the primary load direction. Unidirectional (0°) laminates are extremely strong and stiff in the (0°) direction; however, they are also very weak in the (90°) direction because the load must be carried by the much weaker polymeric matrix.[3] While a high-strength fiber can have a tensile strength of 500 KSI or more, a typical polymeric matrix normally has a tensile strength of only 5-10 KSI [Figure 2.16]. The longitudinal tension and compression loads are carried by the fibers, while the matrix distributes the loads between the fibers in tension and stabilizes and prevents the fibers from buckling in compression. The matrix is also the primary load carrier for inter laminar shear (i.e., shear between the layers) and transverse (90°) tension.[3]

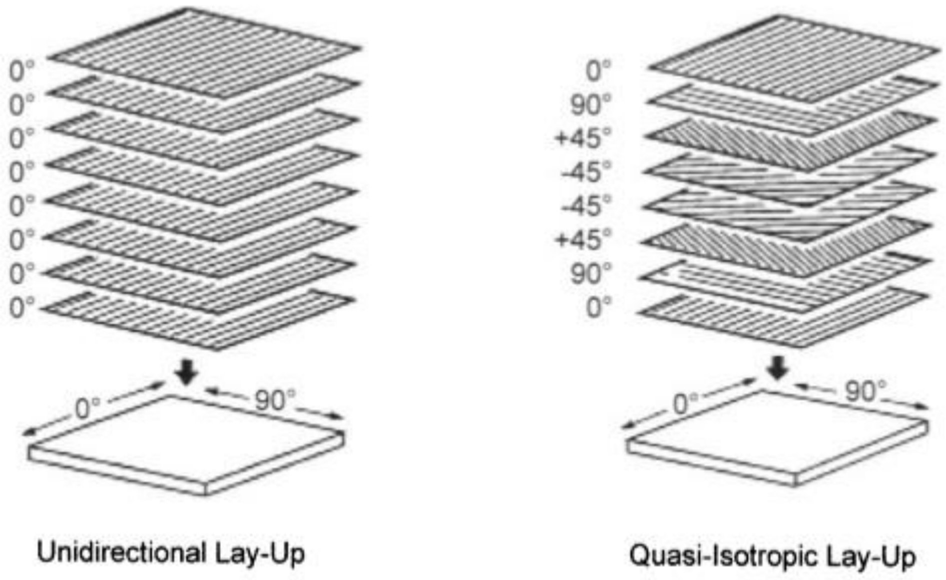


Figure 2.16:Quasi-Isotropic Laminate Lay-Up[3]

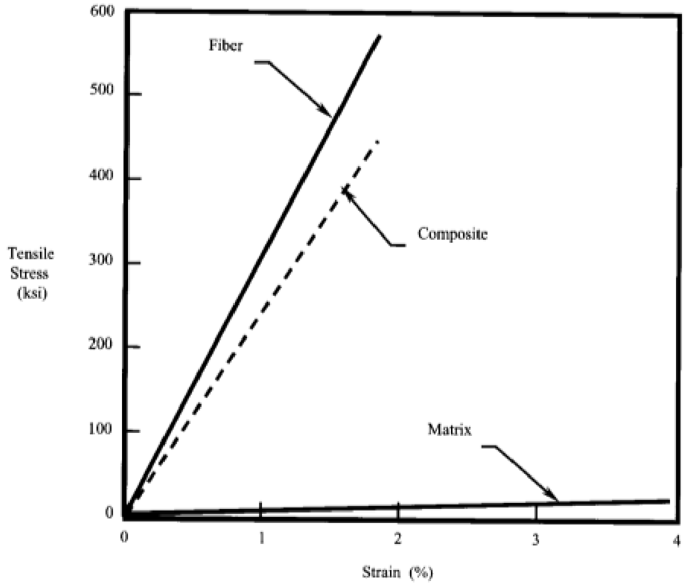


Figure 2.17:Tensile properties of fiber, matrix and composite[3]

2. Sandwich Structure Composite Material

Theory a sandwich construction is a structural panel concept that consists in its simplest form of two relatively thin, parallel face sheets bonded to and separated by a relatively thick, lightweight core. The core supports the face sheets against buckling and resists out-of-plane shear loads. The core must have high shear strength and compression stiffness. Composite sandwich construction is most often fabricated using autoclave cure, press cure, or vacuum bag cure. Skin laminates may be precured and subsequently bonded to core, co-cured to core in one operation, or a combination of the two methods. Examples of honeycomb structure are: wing spoilers, fairings, ailerons, flaps, nacelles, floor boards, and rudders.

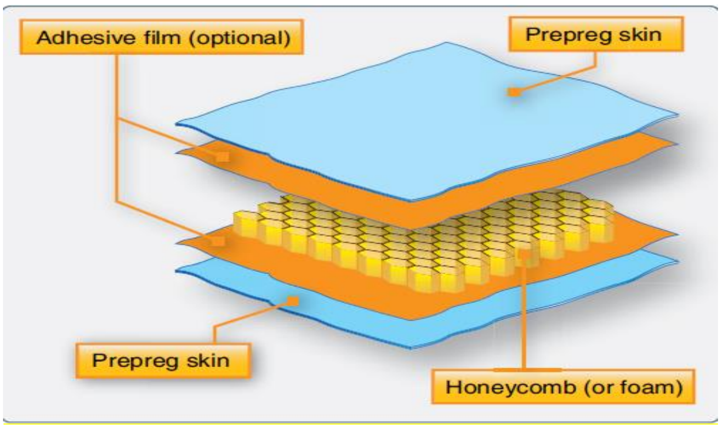
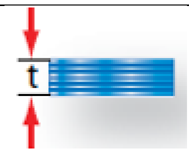
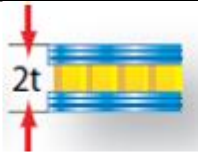
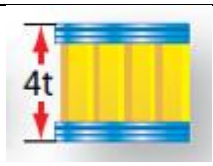


Figure 2.18: honeycomb sandwich construction

i. Properties of Sandwich Construction

Sandwich construction has high bending stiffness at minimal weight in comparison to aluminum and composite laminate construction. Most honeycombs are anisotropic; that is, properties are directional. [Table 2.5] illustrates the advantages of using a honeycomb construction. Increasing the core thickness greatly increases the stiffness of the honeycomb construction, while the weight increase is minimal. Due to the high stiffness of a honeycomb construction, it is not necessary to use external stiffeners, such as stringers and frames. [Table 2.5]

Table 2.5: Strength and stiffness of honeycomb sandwich material

	Solid material	Core thickness	Core thickness 3t
			
Thickness	1.0	7.0	37.0
Flexural strength	1.0	3.5	9.2
Weight	1.0	1.03	1.06

ii. Facing Materials

Most honeycomb structures used in aircraft construction have aluminum, fiberglass, Kevlar®, or carbon fiber face sheets. Carbon fiber face sheets cannot be used with aluminum honeycomb core material, because it causes the aluminum to corrode. Titanium and steel are used for specialty applications in high temperature constructions. The face sheets of many components, such as spoilers and flight controls, are very thin—sometimes only 3 or 4 plies. Field reports have indicated that these face sheets do not have a good impact resistance.

iii. Core Materials

- Honeycomb

Each honeycomb material provides certain properties and has specific benefits. [Figure 2.19] The most common core material used for aircraft honeycomb structures is aramid paper (Nomex® or Korea®). Fiberglass is used for higher strength applications.

a. Kraft paper—relatively low strength, good insulating properties, is available in large quantities, and has a low cost.

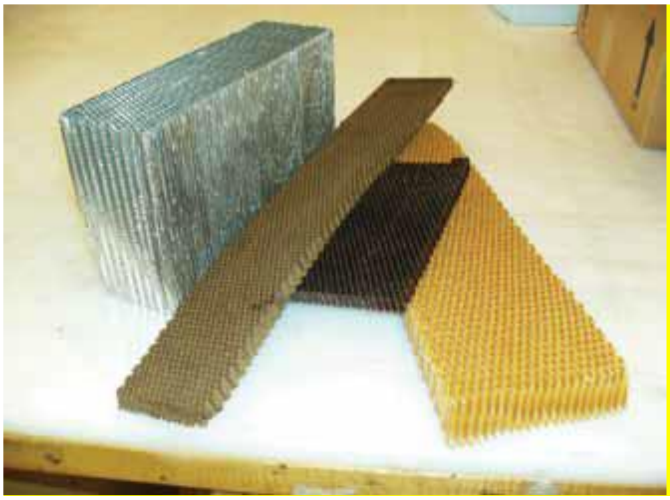


Figure 2.19: honeycomb core material

b. Thermoplastics: good insulating properties, good energy absorption and/or redirection, smooth cell walls, moisture and chemical resistance, are environmentally compatible, aesthetically pleasing and have a relatively low cost.
c. Aluminum: best strength-to-weight ratio and energy absorption, has good heat transfer properties, electromagnetic shielding properties, has smooth, thin cell walls, is machinable, and has a relatively low cost.
d. Steel–good heat transfer properties, electromagnetic shielding properties, and heat resistant.
e. Specialty metals (titanium): relatively high strength to weight ratio, good heat transfer properties, chemical resistance, and heat resistant to very high temperatures.
f. Aramid paper: flame resistant, fire retardant, good insulating properties, low dielectric properties, and good formability.
g. Fiberglass: tailorable shear properties by layup, low dielectric properties, good insulating properties, and good formability.
h. Carbon: good dimensional stability and retention, high-temperature property retention, high stiffness, very low coefficient of thermal expansion, tailorable thermal conductivity, relatively high shear modulus, and very expensive.

i. Ceramics: heat resistant to very high temperatures, good insulating properties, is available in very small cell sizes, and very expensive.

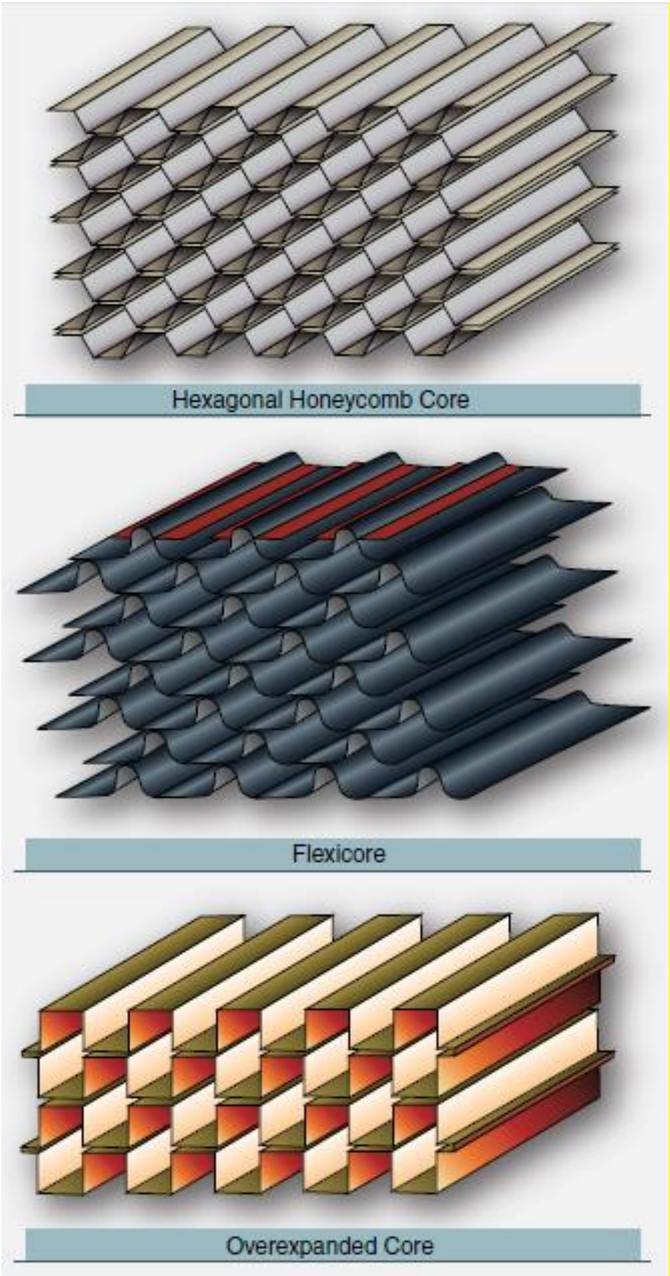


Figure 2.20: honeycomb density

- Foam

Foam cores are used on homebuilt and lighter aircraft to give strength and shape to wing tips, flight controls, fuselage sections, wings, and wing ribs. Foam cores are not commonly used on commercial type aircraft. Foams are typically heavier than honeycomb and not as strong. A variety of foams can be used as core material including:

a. Polystyrene (better known as Styrofoam)—aircraft grade Styrofoam with a tightly closed cell structure and no voids between cells; high compressive strength and good resistance to water penetration; can be cut with a hot wire to make airfoil shapes.
b. Phenolic—very good fire-resistant properties and can have very low density, but relatively low mechanical properties.
c. Polyurethane—used for producing the fuselage, wing tips, and other curved parts of small aircraft; relatively inexpensive, fuel resistant, and compatible with most adhesives; do not use a hot wire to cut polyurethane foam; easily contoured with a large knife and sanding equipment.
d. Polypropylene—used to make airfoil shapes; can be cut with a hot wire; compatible with most adhesives and epoxy resins; not for use with polyester resins, dissolves in fuels and solvents.
e. Polyvinyl chloride (PVC) (Divinycell, Klegecell, and Airex)—a closed cell medium- to high-density foam with high compression strength, durability, and excellent fire resistance; can be vacuum formed to compound shapes and be bent using heat; compatible with polyester, vinyl ester, and epoxy resins.
f. Polymethacrylimide (Rohacell) a closed-cell foam used for lightweight sandwich construction; excellent mechanical properties, high dimensional stability under heat, good solvent resistance, and outstanding creep compression resistance; more expensive than the other types of foams, but has greater mechanical properties.

2.9.3 Particulate Composite Material

Articulate MMCs should be mentioned in this overview because they may have extensive aerospace applications 1° as structural materials. In these composites, aluminum or titanium alloy-matrices are reinforced with ceramic particles, generally silicon carbide or alumina in the micron range. Because reinforcement is not directional as with fiber-reinforced MMCs, properties are essentially isotropic. The specific stiffness of aluminum silicon-carbide particulate MMCs (A1/SICP, where the subscript p refers to particulate) can exceed conventional aluminum alloys by around 50% at a

20% particle volume fraction. For comparison, an MMC with inclusion of silicon-carbide fibers at a similar volume fraction will increase its specific stiffness increased by around 100%.[3]

The primary fabrication techniques are rapid-liquid-metal processes such as squeeze casting or solid-state powder processes based on hot-pressing. Particulate MMCs also have the considerable cost advantage of being formable by conventional metal-working techniques and possibly super-plastic forming and diffusion bonding in the case of titanium-matrix systems. However, because of their high wear resistance, special tools such as diamond-coated drills and diamond-impregnated grinding wheels are required for machining. When fabricated using clean high-grade particles with low porosity and moderate particulate volume fraction, particulate MMCs have high strength, acceptable fracture toughness, and good resistance to fatigue crack propagation. The MMCs also have high stiffness and wear resistance compared with conventional alloys. They are therefore suited to small components requiring high stiffness combined with fatigue and wear resistance.[3]

2.10 Design consideration

General guidelines

Composite structural design should not be attempted without a good working knowledge of the manufacturing limitations applying to composite materials.

Generally, concurrent engineering is practiced whereby designers and manufacturing engineers work toward solutions that satisfy both design intent and production needs.

When specifying lay-ups (laminate ply stacks) and design details, some basic guidelines should be followed:

1. Use balanced laminates to avoid warping
2. Use manufacturing techniques that produce a minimum fiber content of 55% by volume;
3. Use a minimum of 10% of plies in each of the principal directions (0 °, 90 °, 45 °) to provide a minimum acceptable strength in all directions
4. Use a maximum of four adjacent plies in any one direction to avoid splitting on contraction from cure temperature or under load

5. Place + 45 ° plies on the outside surfaces of shear panels to increase resistance to buckling
6. Avoid highly directional laminates in regions around holes or notches because stress concentration factors are significantly higher in this ply lay-up
7. Add ply of woven fiberglass barrier between carbon and aluminum alloy for galvanic protection
8. Drop plies where required progressively in steps with at least 6 mm (0.25 in) landing to improve load redistribution
9. Where possible, cover ply drops with a continuous ply to prevent end-of-ply delamination
10. Maintain three-dimensional edge distance and four-dimensional pitch for mechanical fasteners to maximize bearing strength
11. Where feasible, avoid honeycomb in favor of stiffened construction, because honeycomb is prone to moisture intrusion and is easily damaged
12. Avoid manufacturing techniques that result in poor fiber alignment, because wavy fibers results in reduced stiffness and compression strength
13. Minimize the number of joints by designing large components or sections because joints reduce strength and increase weight and cost
14. Allow for impact type damage (see later discussion); this may vary with risk (e.g., upper horizontal surfaces are at greatest risk).
15. Exploit the non-isotropic properties of the material, where feasible.
16. Ensure that the design reflects the limitations of the manufacturing processes to be used
17. Predict the failure loads and modes for comparison with test data
18. Minimize or exclude the features that expose the notch-sensitivity of the material.
19. Allow for degradation due to the environment.
20. Provide for ready inspection of production defects.
21. Allow for repair in the design.
22. Predict and minimize, by design, out-of-plane loading.
23. Include consideration of residual stresses in the cured laminate when calculating strength.[3]

Figure 2.21: SAFAT 01

2.11 Load distribution

2.11.1 Introduction

Any flying machine, whether it is a manned aircraft or a guided missile, is required to possess a structural strength and stiffness which is adequate to enable it to perform its intended role, and continue to perform it throughout its life. The requirements contain much more than conditions to ensure structural strength and stiffness. The scope ranges over performance, handling, detail design of systems and installations and general operating requirement and procedures. Before an aircraft can be operated it is necessary to demonstrate that it meets all the relevant requirements and any special ones which may be necessary. When the airworthiness authority is satisfied with the general performance and design of the aircraft permission is given to operate it.

The permission may be restrictive in some sense and may not be approved or recognized by other authorities, although it is now normal for design teams to consider all sets of requirements which are likely to be met with.

2.11.2 Loading Action

The aim of loading action analysis is to obtain cases which enable the airframe to be designed and stressed. Many of the methods used for estimating the loads acting on aircraft are complex and involve knowledge of parameters which can only be determined accurately at the later stages of the design.

2.11.2.1 Cause of Loads

Loads result directly from the action of the pilot or the autopilot which is classified as maneuvering loads since they occur as the aircraft carries out its intended role. Cabin pressure differential and the effect of kinetic heating at higher Mach number can be added to the above. The environment in which the aircraft operates. These may be due to such things as atmospheric turbulence, pressure differential, kinetic heating or runway unevenness.

2.11.2.2 Frequency of Loads

Whether the loads be “maneuver” or “environmental” they must be dealt with in two ways for design purposes.

There are the so called “limit load” conditions. The limit load is the actual maximum load of a case which can be experienced in normal conditions, thus a failure due to the application of limit load must be in the extremely remote category. It is in fact the actual maximum load for a maneuver or environmental condition and represents the most severe isolated load intensity.

There is a set of loads of varying magnitude which is experienced by the airframe throughout its life and which arise in any given category Often the majority of these loads are small in comparison with the limit value, but each may damage the structure and it is necessary to ensure that the total accumulated damage due to all of them is within the capability of the structure These loads may arise in a specific way and be of known magnitude and frequency, an example being cabin pressurization.

Many of these loads cannot be dealt with as simply as this and some, especially those due to atmospheric turbulence or runway unevenness are essentially random in nature.

2.11.2.3 Load Factors

Some factors are used for the structure design of the aircraft superimposed on the limit load. They are called the proof factor, the ultimate factor and the load factor. Load factors are applied to limit load cases. There are two factors which are used:

The Proof Factor which is 1.125 for military and 1.0 for civil aircraft. Under proof loading conditions the airframe must not distort permanently more than a small, specified amount, usually (0.1 – 5) % permanent strain depending upon the type of loading. This factor is intended to ensure that the structure will always return to shape after design loading has been applied.

There is an Ultimate Factor which in most instances is 1.5 times the limit load in both military and civil requirements. The Ultimate Factor is a safety factor on the limit load and is intended to cover such items as:

Variation of material and structural properties outside the specified limits, Deterioration in service, Inadequacy of load and stress analysis assumptions, or Possible flight of the aircraft just outside the stated design limitations. During maneuvers or flight through turbulence additional loads are imposed which increase or decrease the net loads on the structure of the aircraft. The amount of this additional load depends upon the severity of the maneuver or the turbulence, and its magnitude is measured in terms of load factor. The load factor is a multiplying factor defines load in terms of weight. The load factors are based on statistical data. Transport aircraft have values 2 to 3 while fighter's aircraft have values 6 to 8.

2.11.3 Certification specifications of very light aircraft CS VLA

CS-VLA is valid for the following type of aircraft:

1. Single engine
2. Max. 2 seats
3. MTOW not more than 750 kg
4. Stalling speed in landing configuration of not more than 45 Km/hr.

2.11.3.1 CS – VLA 301 loads

1. Strength requirements are specified in terms of limit loads (the maximum loads to be expected in service) and ultimate loads (limit loads multiplied by prescribed factors of safety). Unless otherwise provided, prescribed loads are limit loads.

2. Unless otherwise provided, the air, ground, and water loads must be placed in equilibrium with inertia forces, considering each item of mass in the airplane. These loads must be distributed to conservatively approximate or closely represent actual conditions.
3. If deflections under load would significantly change the distribution of external or internal loads, this redistribution must be considered.

2.11.3.2 CS – VLA 303 Safety Factor

Unless otherwise provided, a factor of safety of 1·5 must be used.

CS – VLA 305 Strength and Deformation:

1. The structure must be able to support limit loads without detrimental, permanent deformation. At any load up to limit loads, the deformation may not interfere with safe operation.
2. The structure must be able to support ultimate loads without failure for at least three seconds However, when proof of strength is shown by dynamic tests simulating actual load conditions, the three second limit does not apply.

2.11.3.3 CS – VLA 307 Proof of Structure

Compliance with the strength and deformation requirements of CS-VLA 305 must be shown for each critical load condition, Structural analysis may be used only if the structure conforms to those for which experience has shown this method to be reliable. In other cases, substantiating load tests must be made. Dynamic tests, including structural flight tests, are acceptable if the design load conditions have been simulated.

2.11.3.4 CS – VLA 455 Ailerons

The ailerons must be designed for the loads to which they are subjected:

1. In the neutral position during symmetrical flight conditions; and
2. By the following deflections (except as limited by pilot effort), during unsymmetrical flight conditions:
    a) Sudden maximum displacement of the aileron control at V_A. Suitable allowance may be made for control system deflections.
    b) Sufficient deflection at V_C
    c) Sufficient deflection at V_D
3. The average loading.

2.12 Critical literature review

Composite material used in aircraft structure by AHMED ELSAYED and RUDWAN ALSADIG.

a. Methodology is study to composite, data collection and application methods designed and tested.
b. Methods CATIA, ANSYS and EXCEL.

The relevant report uses **CATIA**, **ANSYS** and **EXCEL** and we use **CATIA V5R18, PATRAN 2012** and **NASTRAN** because:

a. The **NASTRAN** high accuracy and the simulation equal the reality
b. **ANSYS** it’s easy to understand and low accuracy
c. **PATRAN** is the most widely used pre/post processing software for (**FEA**) and provides a rich set of tools.

3 Chapter Three: Manufacturing Process

3.1 Manufacturing of Composites

1. Open Mold Processes- some of the original FRP manual procedures for laying resins and fibers onto forms
2. Closed Mold Processes- much the same as those used in plastic molding
3. Filament Winding- continuous filaments are dipped in liquid resin and wrapped around a rotating mandrel, producing a rigid, hollow, cylindrical shape.
4. Pultrusion Processes- like extrusion only adapted to include continuous fiber reinforcement
5. Other PMC Shaping Processes

3.2 Classification of FRP Processes

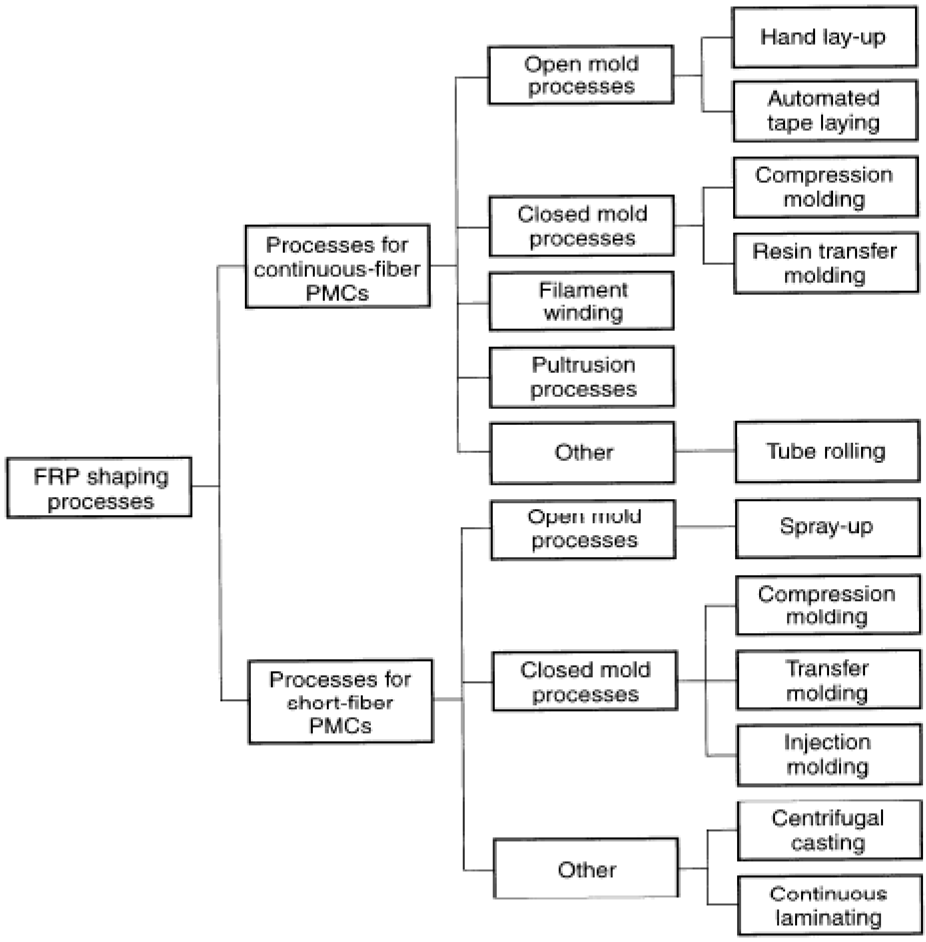


Figure 3.1: FRP processes

3.3 Prepregs

Fibers impregnated with partially cured TS resins to facilitate shape processing. Available as tapes or cross-plied sheets or fabrics curing is completed during and/or after shaping

Advantage: prepregs are fabricated with continuous filaments rather than chopped random fibers, thus increasing strength and modulus

3.3.1 Open Mold FRP Processes

1. Hand lay-up
2. Spray-up
3. Vacuum Bagging – uses hand-lay-up, uses atmospheric pressure to compact laminate.
4. Automated tape-laying machines. The differences are in the methods of applying the laminations to the mold, alternative curing techniques, and other differences

3.3.1.1 Hand Lay-Up

Hand lay–up, or contact molding, is the oldest and simplest way of making fiberglass–resin composites. Applications are standard wind turbine blades, boats, etc.)[9]

Figure 3.2: hand lay-up

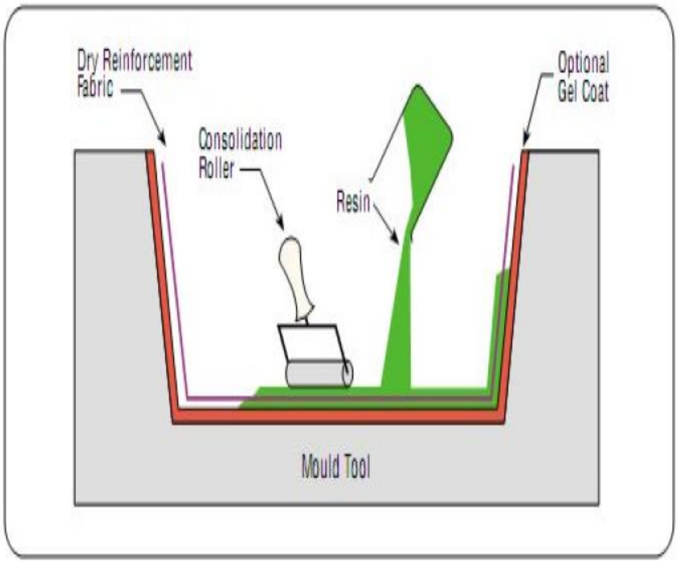


Figure 3.3: hand lay-up schematic

Hand lay-up: (1) mold is treated with mold release agent; (2) thin gel coat (resin) is applied, to the outside surface of molding; (3) when gel coat has partially set, layers of resin and fiber are applied, the fiber is in the form of mat or cloth; each layer is rolled to impregnate the fiber with resin and remove air; (4) part is cured; (5) fully hardened part is removed from mold.[9]

3.3.1.2 Spray-Up Method

In Spray–up process, chopped fibers and resins are sprayed simultaneously into or onto the mold. Applications are lightly loaded structural panels, e.g. caravan bodies, truck fairings, bathtubs, small boats, etc.[9]

The vacuum–bag process was developed for making a variety of components, including relatively large parts with complex shapes. Applications are large raising boats, racecar components, etc.[9]

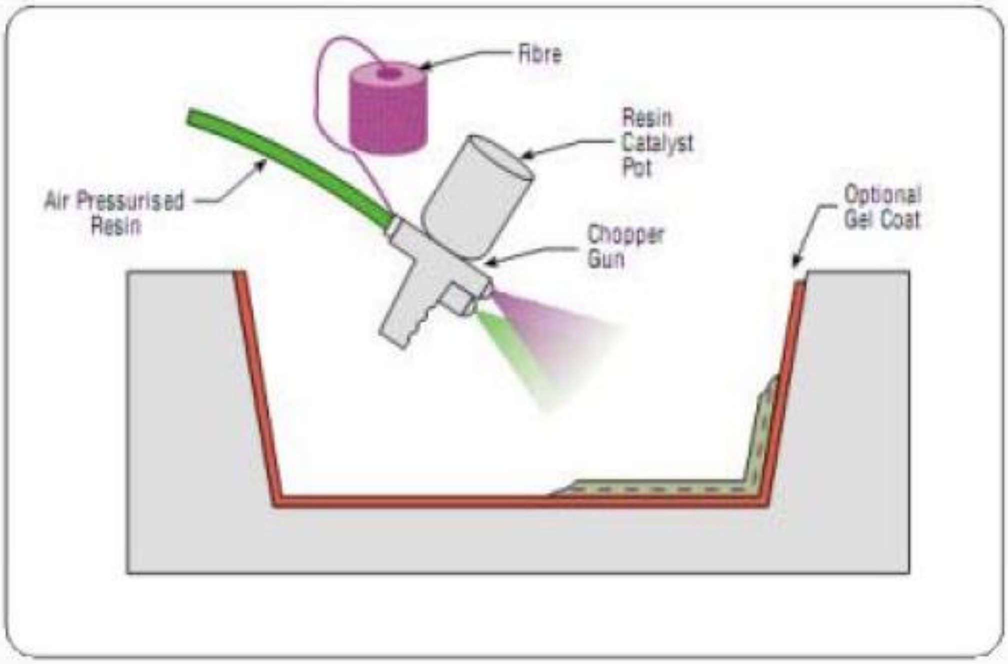


Figure 3.4:Vacuum-Bag Molding

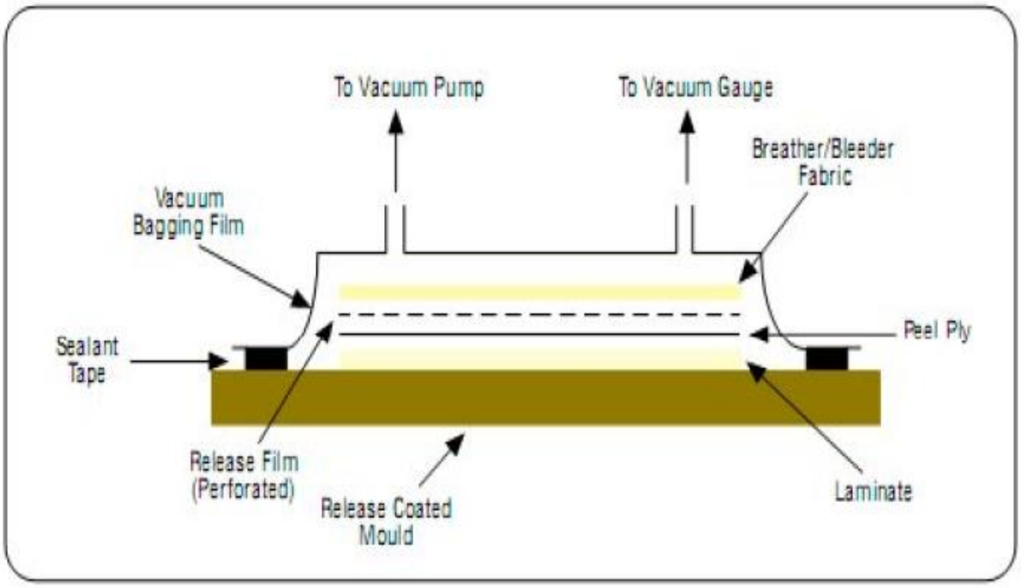


Figure 3.5: Vacuum Bagging Schematic

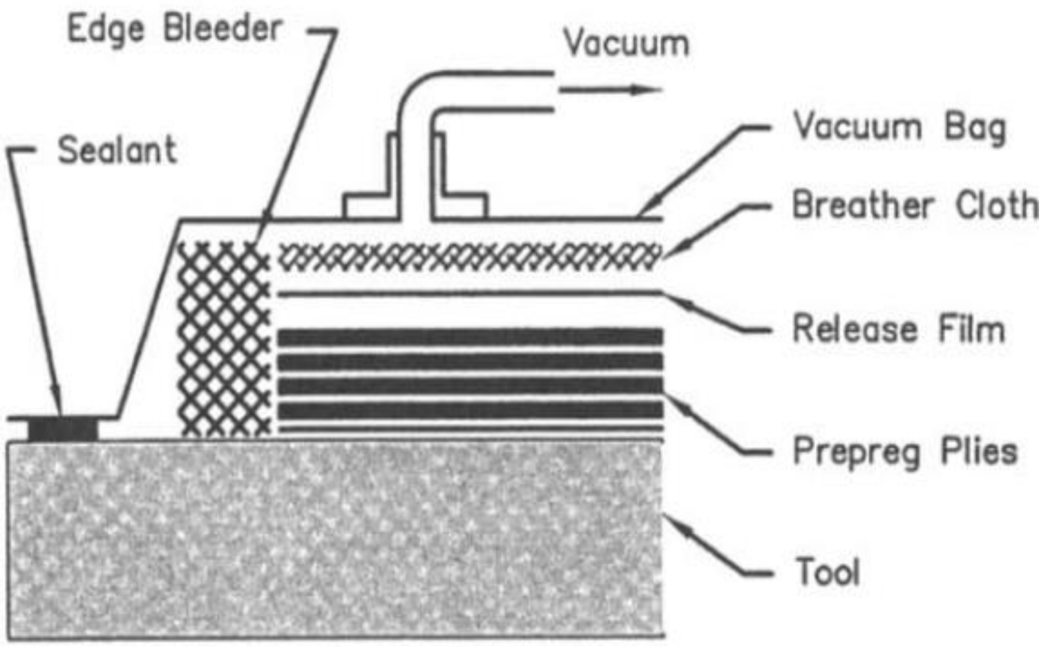


Figure 3.6: Lay-up sequence for bagging operation

Use atmospheric pressure to suck air from under vacuum bag, to compact composite layers down and make a high-quality laminate.

Layers from bottom include: mold, mold release, composite, peel-ply, breather cloth, vacuum bag, also need vacuum valve, sealing tape.[9]

3.3.1.3 Pressure-Bag Molding

Pressure–bag process is virtually a mirror image of vacuum–bag molding. Applications are sonar domes, antenna housings, aircraft fairings, etc.[9]

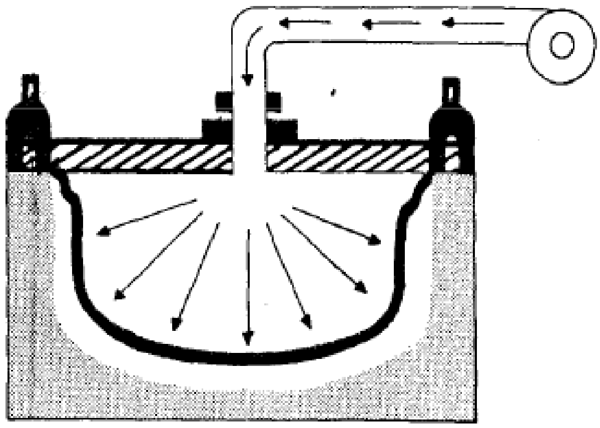


Figure 3.7: Pressure-Bag schematic

3.3.1.4 Thermal Expansion Molding

Prepreg layers are wrapped around rubber blocks and then placed in a metal mold.

As the entire assembly is heated, the rubber expands more than the metal, putting pressure on the laminate.[9]

Complex shapes can be made reducing the need for later joining and fastening operations[9]

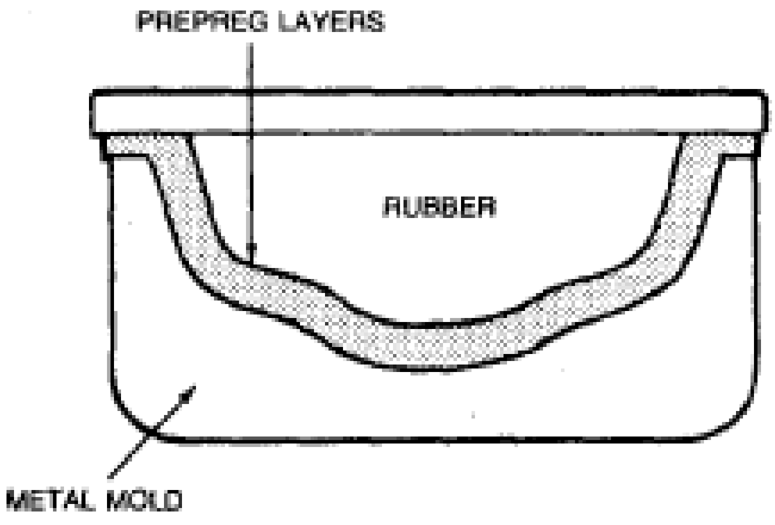


Figure 3.8: Thermal Expansion Molding

3.3.1.5 Filament Winding

Resin-impregnated continuous fibers are wrapped around a rotating mandrel that has the internal shape of the desired FRP product; the resin is then cured, and the mandrel removed[9]

The fiber roving is pulled through a resin bath immediately before being wound in a helical pattern onto the mandrel[9]

The operation is repeated to form additional layers, each having a crisscross pattern with the previous, until the desired part thickness has been obtained.

The advantages of manufacturing by Filament Winding are

1. Highly automated
2. Low manufacturing costs if high throughput– e.g., Glass fiber pipe, sailboard masts.

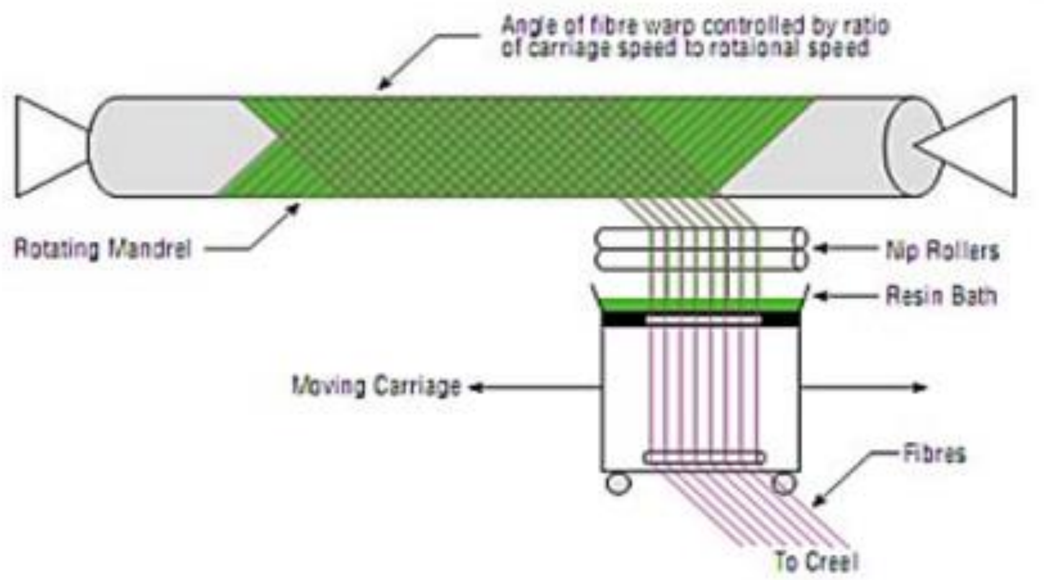


Figure 3.9: filament winding schematic

3.3.1.6 Pultrusion-Process

Continuous fiber roving is dipped into a resin bath and pulled through a shaping die where the impregnated resin cures. The sections produced are reinforced throughout their length by continuous fibers.[9]

Like extrusion, the pieces have a constant cross section, whose profile is determined by the shape of the die opening. The cured product is cut into long straight sections.

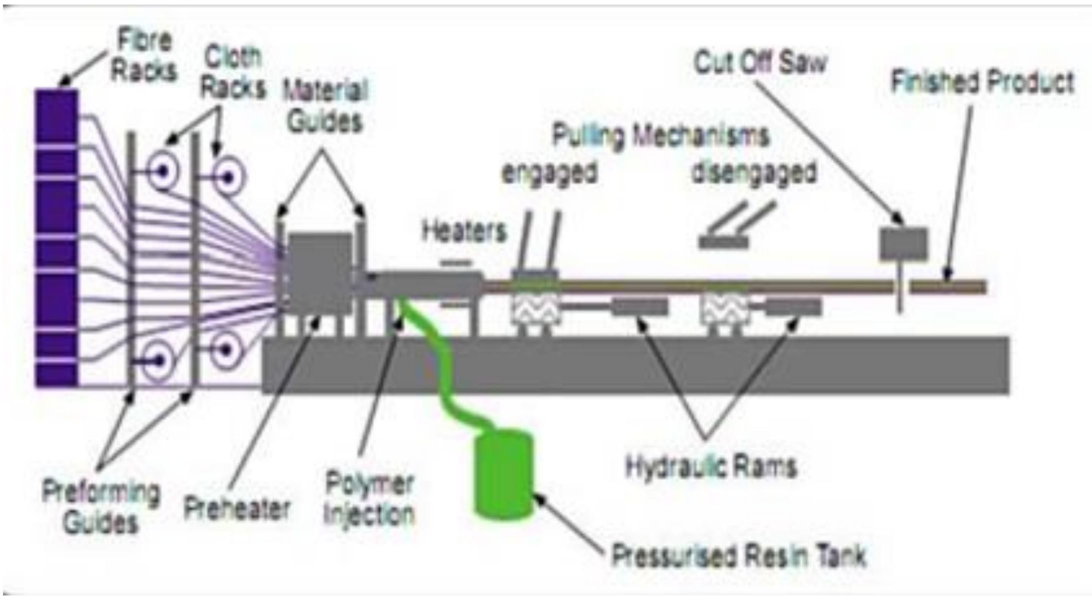


Figure 3.10: Pultrusion schematic.

3.3.2 Closed Mold Processes

Performed in molds consisting of two sections that open and close each molding cycle

Tooling cost is more than twice the cost of a comparable open mold due to the more complex equipment required in these processes.[9]

Advantages of a closed mold are: (1) good finish on all part surfaces, (2) higher production rates, (3) closer control over tolerances, and (4) more complex three-dimensional shapes are possible.[9]

There are three classes of classification of closed mold processes based on their counterparts in conventional plastic molding

3.3.2.1 Compression Molding

Molding compound paced in matched die molding, pressure and heat applied by molding.[10]

Advantage: increased production rat.

Disadvantage: expensive tools

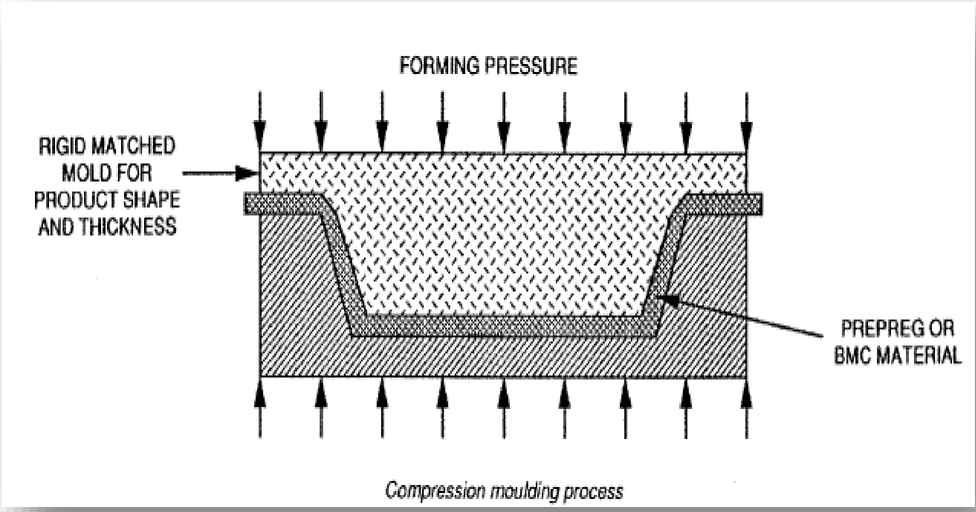


Figure 3.11: compression molding.[10]

3.3.2.2 Transfer Molding

Reinforcement placed between two parts of mold, resin injected in mold and heat applied.[10]

The advantages of resin transfer process molding are accurate in molded surface and Co-curing reduced part count, while the disadvantages are low fiber volume and material property variability throughout part

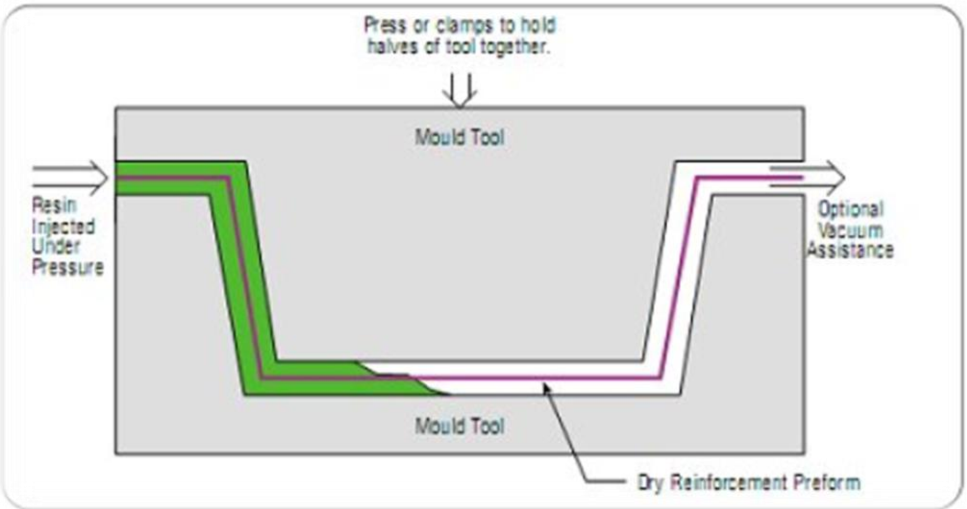


Figure 3.12: resin transfer molding[10]

The terminology is often different when polymer matrix composites are molded.

3.4 Material Selection

The first step in predesign is to select the material (fiber, resin… etc.)

Material selection is usually based on several requirements: environment, stiffness, strength, coefficient of thermal expansion, weight, cost and machine ability.[10]

1. Environment

Temperature

Table 3.1: maximum use temperature.[10]

Material system	Maximum use temperature (degrees F)
Epoxy	260
Bismaleimide	450
Polyimide	550
Experimental (PMR)	700
Aluminum matrix	500
Titanium matrix	1000
Silicon carbide matrix	2500
Coated carbon-carbon	3500

Moisture: organic matrix systems absorb some water and swell. Kevlar fiber absorbs water.[10]

Corrosion fiberglass acts with aluminum and steel (okay for titanium)[10]

2. Stiffness

Buckling or deflection- critical components

Table 3.2: long modulus E11 (MSI).[10]

Composite system	Long. Modulus E11 (MSI)
E-glass/epoxy	5.5
S2-glass/epoxy	7.1
Kevlar/epoxy	11.0
Graphite/epoxy	20.4
HM graphite/epoxy	31.2
UHM graphite/epoxy	45.0

Notes:

1) All systems are UD tape for comparison
2) High modulus
3) Ultra-high modulus

3. Strength

Table 3.3: strength of composite systems.[10]

Composite system	Ftu (KSI)	Fcu (KSI)	Fsu (KSI)
E-glass/epoxy	175	85	9
S2-glass/epoxy	221	124	12
Kevlar/epoxy	200	40	6
Graphite/epoxy	210	172	13

Note: all systems are UD tape for comparison.

4. Coefficient of thermal expansion (CTE)

Table 3.4: Composite system Long. CTE (micro-in/in/F).[10]

Composite system	Long. CTE (micro-in/in/F)
E-glass/epoxy	3.3
S2-glass/epoxy	2.2
Kevlar/epoxy	-2.2
Graphite/epoxy	-0.3

Note: all systems are UD tape for comparison.

5. Weight

Table 3.5: density of composite systems.[10]

Composite system	Density (pci)
E-glass/epoxy	0.075
S2-glass/epoxy	0.072
Kevlar/epoxy	0.049
Graphite/epoxy	0.055

6. Cost

Table 3.6: cost of composite system.[10]

Composite system	Cost ($/lb)
E-glass/epoxy	9
S2-glass/epoxy	13
Kevlar/epoxy	27
Graphite/epoxy	30

Note: cost numbers shown for fabric and epoxy with RTM process.

7. Machineability

Kevlar/epoxy leaves rough surface.

Glass and graphite machine well.[10]

4 Chapter Four: Calculation

4.1 Aircraft Parameters

Table 4.1: SAFAT 01 parameters

Total weight	6867 N
Service ceiling	3 Km
Range	1120 Km
Fuel tank capacity	120 liters
Aileron deflections	15 deg down, 20 deg up
Flap deflections	20 deg, 45 deg
Engine horsepower	150 hp
Wing span	10.25 m
Wing airfoil	USA 35B
Aspect ratio	6.944
Wing area	159.7 sq.ft.
Flap area	18.8 sq.ft.
Take off distance	50 m
Stall speed	40 Km/ hr-with flap 50 Km/ hr
Rate of climb	2 m/s

4.2 V-n Diagram

Figure (4.1) shows the V-n diagram for SAFAT01, this flight envelope shows the critical design points for the aircraft, the gust load is included and has no significant effect, the envelope is constructed during the design phase of SAFAT 01 and a design report from ARDC (AERONAUTICAL RESEARCH AND DEVELOPMENT CENTER) is available there.

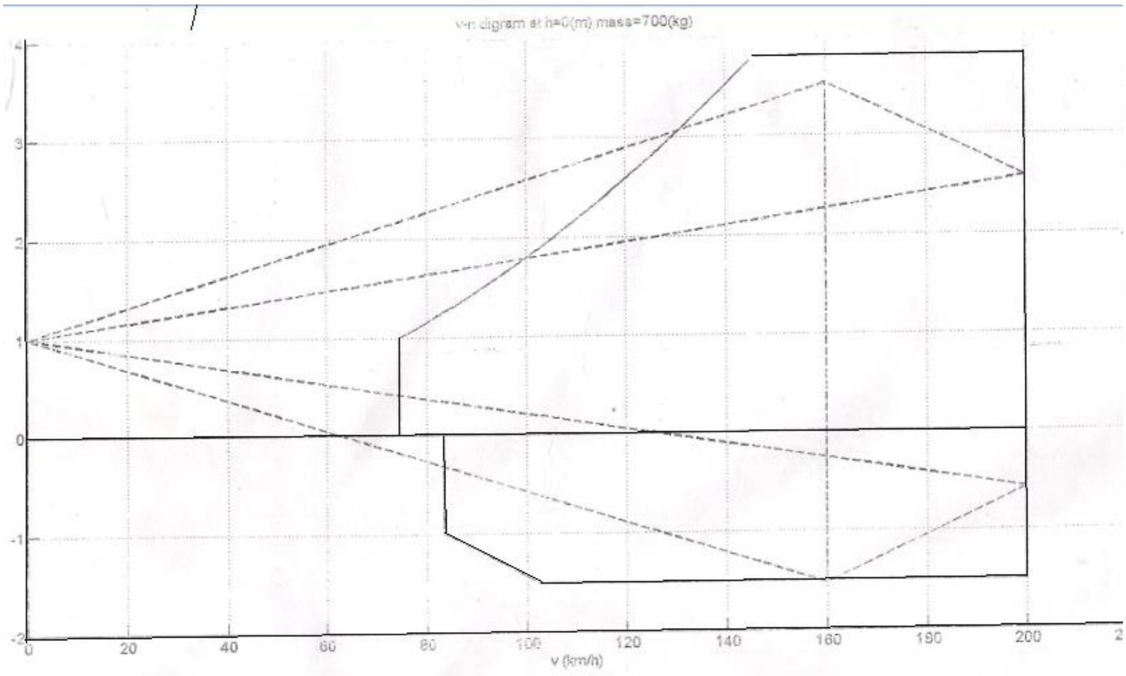


Figure 4.1:v-n diagram

4.3 Wing Layout

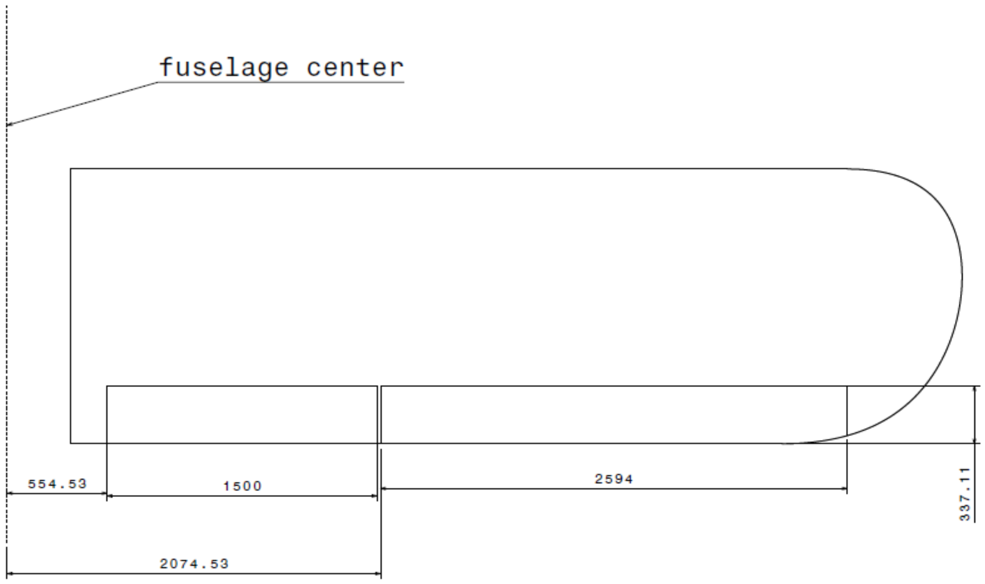


Figure 4.2:wing layout

4.4 Aileron Dimensions

Table 4.2: aileron dimensions

Aileron area	0.874463.34 m^2	9.41295 sq.ft.
Aileron length	2.594 m	8.510 ft
Aileron chord	0.33711 m	1.106 ft
Aileron hinge position from leading edge	0.866 m	0.276 m

4.5 Regulations

The following load analysis is based on the “Certification Specifications for Very Light Aircraft” issued by the European Aviation Safety Agency.

4.6 Aileron Load Analysis (c) (1)

The average limit loading of the control surfaces can be calculated according to CS-VL.

Simplified limit surface distributions:

$$Wail = 0.095 * n1 * \frac{w}{s} \quad \text{………………………..} \quad (4.1)$$

Where W_{ail} is the average surface loading

N1 is the maximum load factor

W is the total weight of the of the aircraft

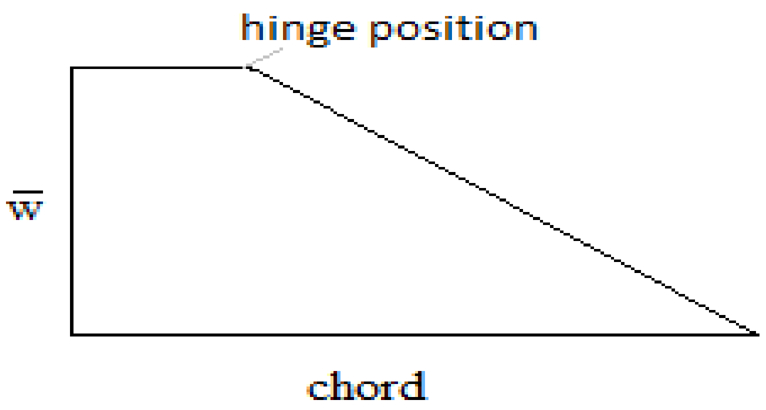


Figure 4.3: aileron Chord wise load distribution

The calculation results are summarized in [Table 4.3] and [Table 4.4]

Table 4.3: aileron load.

Load case	Load up and down position
Average surface loading W	167.097 N/m^2
Limit load (chord wise)	35.1728 N/m
Total limit load	91.23823 N
Ultimate load (chord wise)	52.75919 N
Total ultimate load	136.8574 N

Table 4.4: aileron load calculation results

Load case	Load up and down position
Max shear load	0.00684287 N
Max bending moment	19.72267185 Nm
Max torsion	1.259 Nm

4.7 Geometric Model

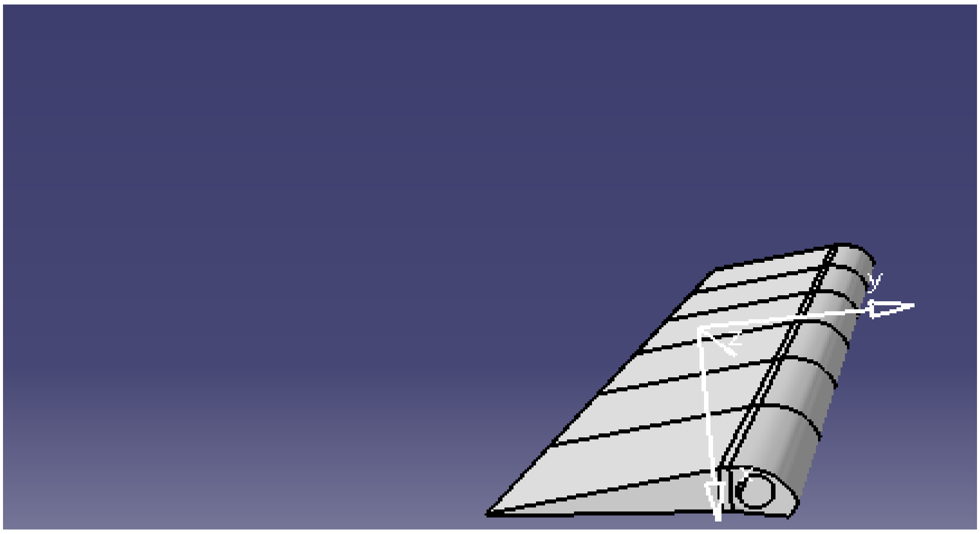


Figure 4.4: CATIA drawing model

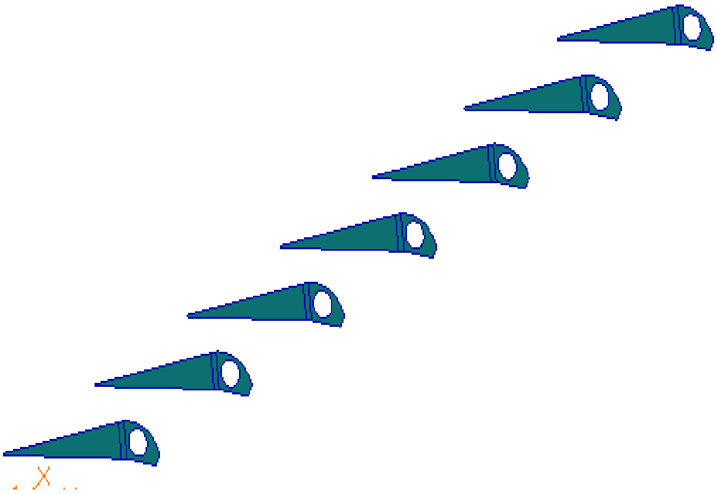


Figure 4.5: rib model

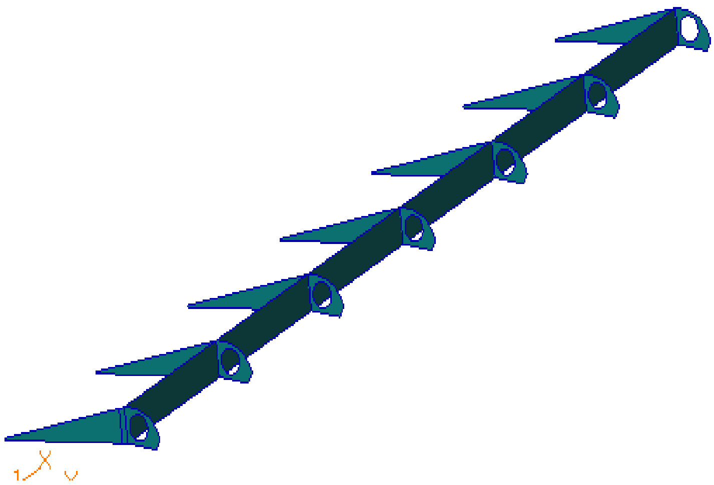


Figure 4.6: rib and spar model

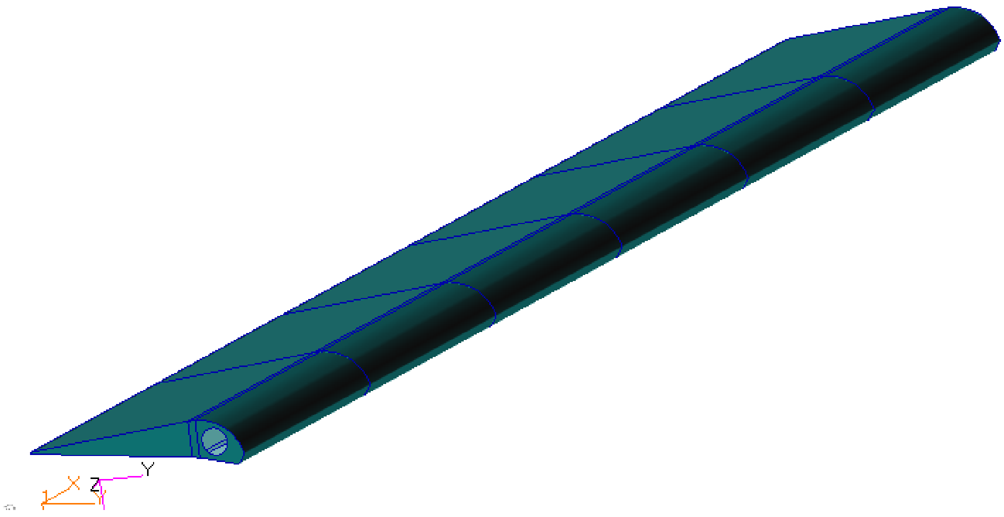


Figure 4.7: skin and ribs model.

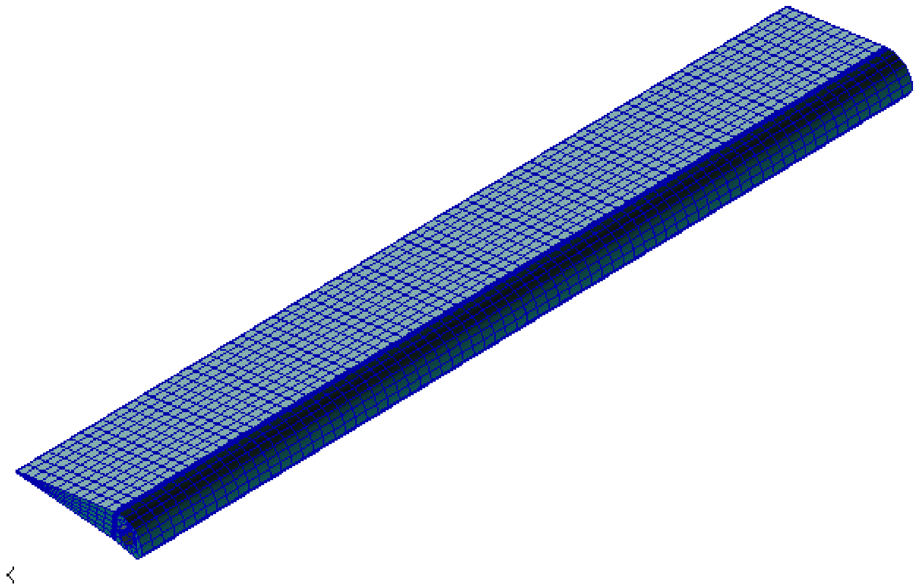


Figure 4.8: mesh model

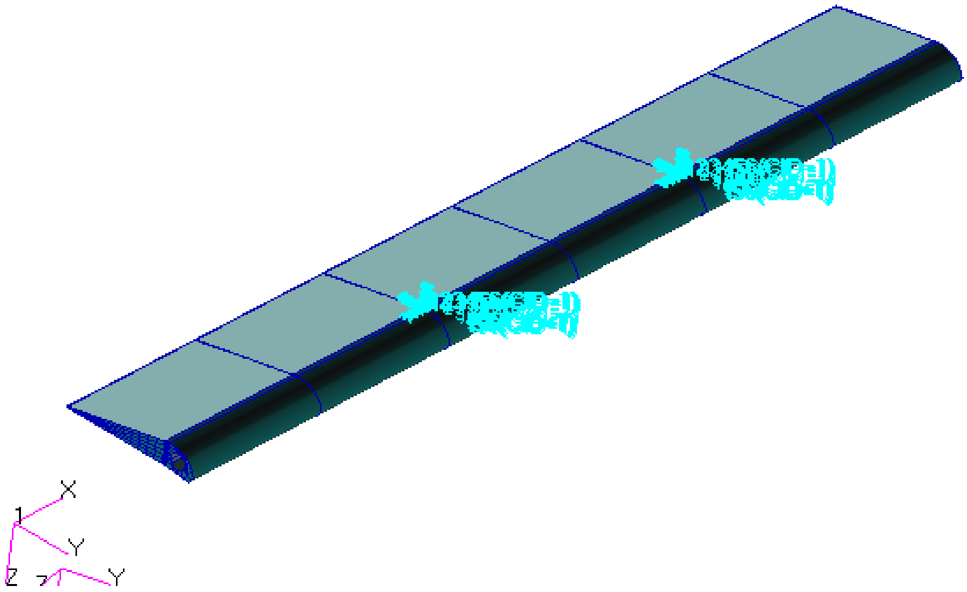


Figure 4.9: fixed model

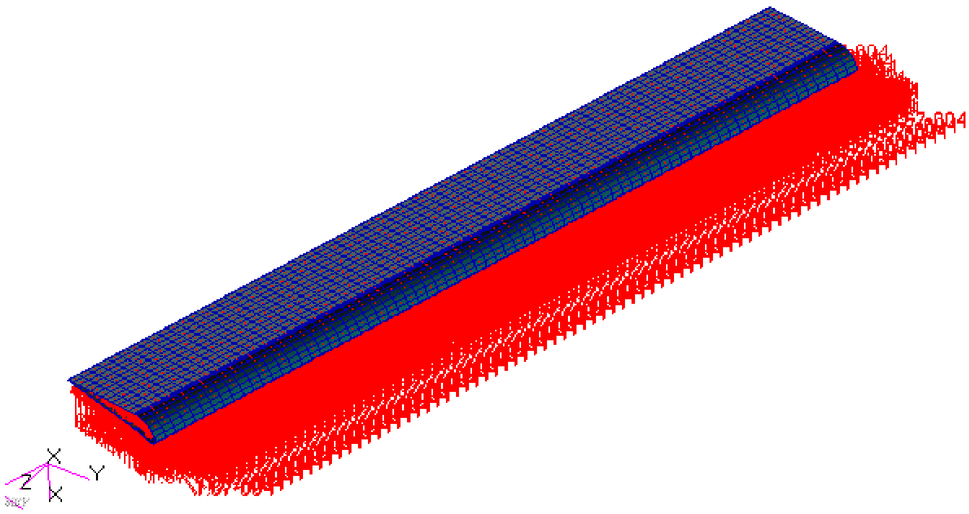


Figure 4.10:pressure load distribution on the aileron model

Mechanical properties of unidirectional ply are following

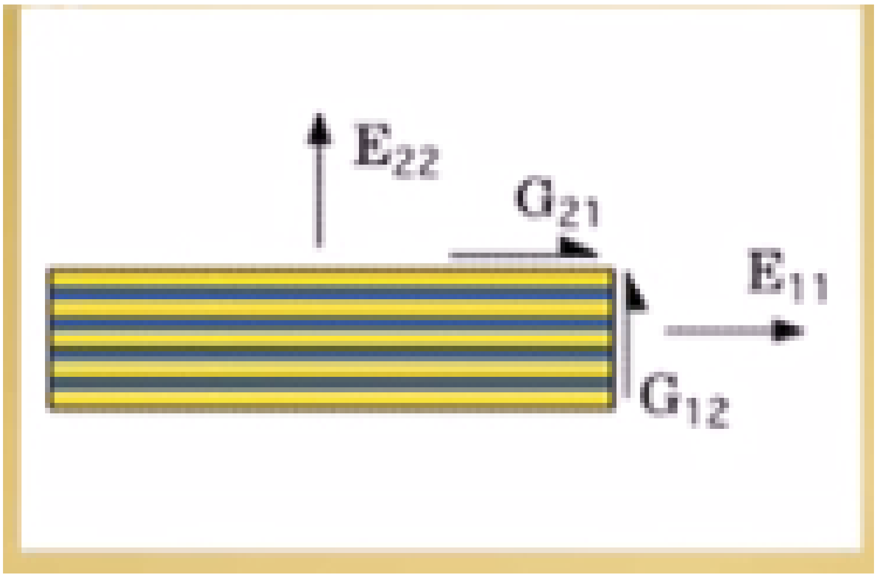


Figure 4.11: mechanical properties unidirectional ply

Ply level property may also be estimated by rule of mixtures for unidirectional ply:

$$E_{11} = E_f * \nu_f + E_m * \nu_m \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots.. (4.2)$$

$$E_{22} = (E_f * E_m)/(E_f * (E_m - E_f) + \mathrm{E}_f \dots\dots\dots\dots\dots..\dots. (4.3)$$

$$G_{12} = \left(\frac{1}{\frac{\nu_f}{G_f}} + \frac{1-\nu_f}{G_f} \right) \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots.. (4.4)$$

$$\nu_{12}, \nu_{21} = \nu_f V_f + \nu_m(1-\nu_f) \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots.. (4.5)$$

$$F_{11} = F_f \nu_f + F_m(1-\nu_f) \dots\dots\dots\dots\dots\dots\dots\dots\dots..\dots\dots\dots.. (4.6)$$

$E_m, G_m, E_m, \nu_m \equiv$ Properties of matrix material

$E_f, G_f, E_f, \nu_f \equiv$ Properties of material

$F_{11} \equiv$ Strength in fiber direction

$F_{22} \equiv$ Strength in transverse direction.

Carbon/epoxy unidirectional continuous fiber composite with fiber volume ration, $V_f = 0.6$ & the following fiber & matrix properties.

Table 4.5 : tensile strength, tensile modulus, shear modulus, specific gravity and poison ratio of carbon fiber and epoxy resin.

	Tensile modulus GPa	Tensile strength (GPa)	Shear modulus (GPa)	Specific gravity (gm/cc)	Poison ratio
Carbon fiber	230	3.2	50	1.75	0.3
Epoxy	4.5	0.13	1.6	12	0.4

Calculate$E_{11}, E_{22}, G_{12}, \nu_{12}$

$$E_{11} = E_f * \nu_f + E_m * \nu_m = 230*0.3+4.5*0.4=139\text{Gpa}$$

$$E_{22} = (E_f * E_m)/(E_f * (E_m - E_f) + E_f) = \frac{230*4.5}{(0.6*(4.5-230)+230)} = 10\text{Gpa}$$

$$G_{12} = \left(\frac{1}{\frac{\nu_f}{G_f}} + \frac{1-\nu_f}{G_f}\right) = \left(\frac{1}{\frac{0.6}{50}} + \left(\frac{1-0.6}{1.6}\right)\right) = 3.83\text{Gpa}$$

$$\nu_{21} = \nu_f V_f + \nu_m(1-\nu_f) = 0.3*0.6+0.4(1-0.6) = 0.34$$

Table 4.6: $E_{11},, E_{22}, G_{12}$ *and* ν_{12}values for each carbon/epoxy and aluminum.

Engg. Constants	**Carbon/Epoxy**	**Aluminum**
E_{11}	139GPa	70GPa
E_{22}	10GPa	70GPa
G_{12}	3.8GPa	26.92GPa
ν_{12}	0.34	0.3

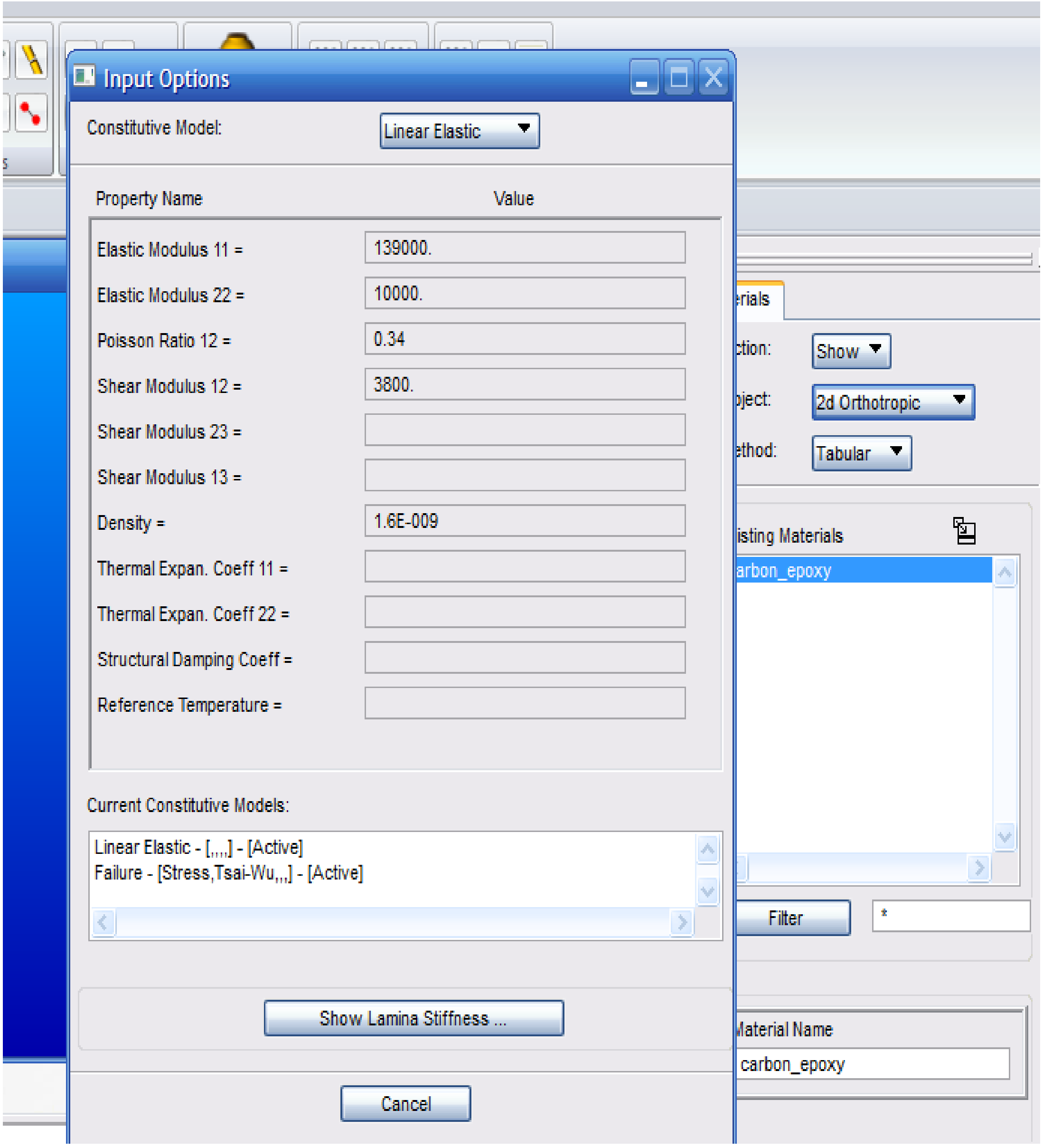


Figure 4.12: input data.

Stress –Strain law for a single ply in the material axes

Unidirectional laminates

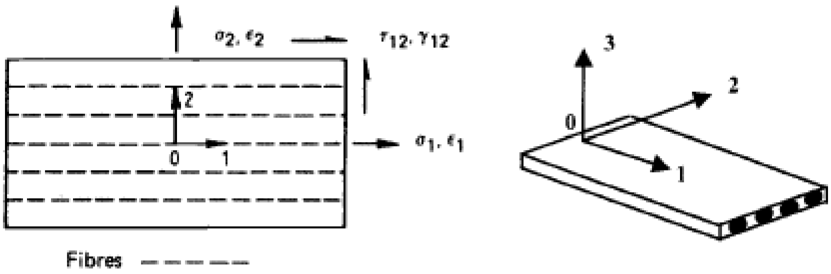


Figure 4.13: Material axes for a single ply.

$$\begin{bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{bmatrix} = \begin{bmatrix} \frac{1}{E_{11}} & \frac{-\nu_{21}}{E_{22}} & 0 \\ \frac{-\nu_{12}}{E_{11}} & \frac{1}{E_{22}} & 0 \\ 0 & 0 & \frac{1}{G_{12}} \end{bmatrix} \begin{bmatrix} \sigma_1 \\ \sigma_1 \\ \tau_{12} \end{bmatrix} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots . \, (4.7)$$

Where:

$\mathrm{E}_{11}, \mathrm{E}_{22}$ ≡ Young's moduli in the 1 and 2 directions, respectively.

ν_{12} ≡ Poison's ratio governing the contraction in the 2 directions for a tension in the 1 direction.

ν_{21} ≡ Poison's ratio governing the contraction in the 1 direction for a tension in the 2direction.

G_{12} ≡ Shear modulus.

$\varepsilon_x, \varepsilon_y, \gamma_{12}$ ≡ Strains at any point

σ_1 ≡ Longitudinal strength (both tensile and compressive).

σ_2 ≡ Transverse strength (both tensile and compressive).

τ_{12} ≡ Shear strength

$$\frac{\nu_{12}}{E_{11}} = \frac{\nu_{21}}{E_{22}} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots . . (4.8)$$

For much of the following analysis, it is more convenient to deal with the inverse from of equation (4.1), namely

$$\begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \gamma_{12} \end{bmatrix} = \begin{bmatrix} Q_{11} & Q_{12} & 0 \\ Q_{12} & Q_{22} & 0 \\ 0 & 0 & Q_{66} \end{bmatrix} \begin{bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \tau_{12} \end{bmatrix} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots . . \, (4.9)$$

Where the Q_{ij} (0°), commonly termed the reduced stiffness coefficients, given by

$$Q_{11}\,(0^\circ) = \frac{E_{11}}{1 - \nu_{21}\nu_{12}} \ldots\ldots\ldots\ldots(4.10)$$

$$Q_{22}\,(0^\circ) = \frac{E_{22}}{1 - \nu_{21}\nu_{12}} \ldots\ldots\ldots\ldots(4.11)$$

$$Q_{12}\,(0^\circ) = \frac{\nu_{21}E_{11}}{1 - \nu_{21}\nu_{12}} \ldots\ldots\ldots\ldots(4.12)$$

$$Q_{66}\,(0^\circ) = G_{12} \ldots\ldots\ldots\ldots,,\ldots(4.13)$$

Off –Axis laminates

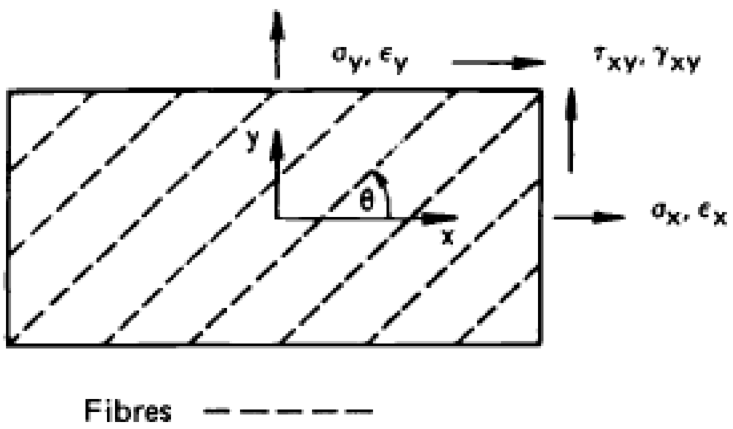


Figure 4.14: Laminate axes for a single ply.

If the stresses in the laminate axes are denoted by σ_x, σ_y and τ_{xy}, then these are related to the stresses referred to the usual transformation equation,

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix} = \begin{bmatrix} C^2 & S^2 & -2CS \\ S^2 & C^2 & 2CS \\ CS & -CS & C^2 - S^2 \end{bmatrix} \begin{bmatrix} \sigma_1 \\ \sigma_2 \\ \tau_{12} \end{bmatrix} \ldots\ldots\ldots\ldots(4.14)$$

Where

$C \equiv \cos\theta$ and $S \equiv \sin\theta$.

Also, the strains in the material axes are related to those in the laminate axes, namely ε_x , ε_y and γ_{xy} by what is essentially the strain transformation:

$$\begin{bmatrix} \varepsilon_1 \\ \varepsilon_2 \\ \gamma_{12} \end{bmatrix} = \begin{bmatrix} C^2 & S^2 & CS \\ S^2 & C^2 & -CS \\ -2CS & 2CS & C^2 - S^2 \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{bmatrix} \dots\dots\dots\dots\dots\dots\dots.\dots (4.15)$$

The stress-strain law in the laminate axes has form

$$\begin{bmatrix} \sigma_x \\ \sigma_y \\ \tau_{xy} \end{bmatrix} = \begin{bmatrix} Q_{xx}\,(\theta^\circ) & Q_{xy}\,(\theta^\circ) & Q_{xs}\,(\theta^\circ) \\ Q_{xy}(\theta^\circ) & Q_{yy}\,(\theta^\circ) & Q_{ys}\,(\theta^\circ) \\ Q_{xs}\,(\theta^\circ) & Q_{ys}\,(\theta^\circ) & Q_{ss}\,(\theta^\circ) \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \gamma_{xy} \end{bmatrix} \dots\dots\dots\dots.\dots (4.16)$$

Where the$Q_{ij}\,(\theta^\circ)$are related to the $Q_{ij}\,(0^\circ)$ by the following equations:

$$\begin{bmatrix} Q_{xx}\,(\theta^\circ) \\ Q_{xy}\,(\theta^\circ) \\ Q_{yy}\,(\theta^\circ) \\ Q_{xs}\,(\theta^\circ) \\ Q_{ys}\,(\theta^\circ) \\ Q_{ss}\,(\theta^\circ) \end{bmatrix} = \begin{bmatrix} C^4 & 2C^2S^2 & S^4 & 4C^2S^2 \\ S^2C^2 & C^4 + S^4 & C^2S^2 & -4C^2S^2 \\ S^4 & 2C^2S^2 & C^4 & 4C^2S^2 \\ SC^3 & -CS(C^2 - S^2) & -CS^3 & -2CS(C^2 - S^2) \\ S^3C & CS(C^2 - S^2) & -SC^3 & 2CS(C^2 - S^2) \\ C^2S^2 & -2C^2S^2 & C^2S^2 & (C^2 - S^2)^2 \end{bmatrix} \begin{Bmatrix} Q_{11}\,(0^\circ) \\ Q_{12}\,(0^\circ) \\ Q_{22}\,(0^\circ) \\ Q_{66}\,(0^\circ) \end{Bmatrix} (4.17)$$

With

$$Q_{xx}\,(\theta^\circ) = C^4 Q_{11}\,(0^\circ) + S^4 Q_{22}\,(0^\circ) + \left(2C^2S^2 Q_{12}\,(0^\circ) + 4C^2S^2 Q_{66}\,(0^\circ)\right)$$

$$Q_{xx}\,(\theta^\circ) = \quad C^4 Q_{11}\,(0^\circ) + 2C^2S^2(Q_{12}\,(0^\circ) + 2Q_{66}\,(0^\circ)) \dots\dots\dots..(4.18)$$

$$Q_{xy}\,(\theta^\circ) = Q_{y\mathrm{x}}(\theta^\circ)$$

$$= C^2S^2\left(Q_{11}\,(0^\circ) + Q_{22}\,(0^\circ) - 4Q_{66}\,(0^\circ)\right) + \left((C^4 + S^4)Q_{12}\,(0^\circ)\right) \dots.(4.19)$$

Calculate the $A^{*}{}_{ij}$ from

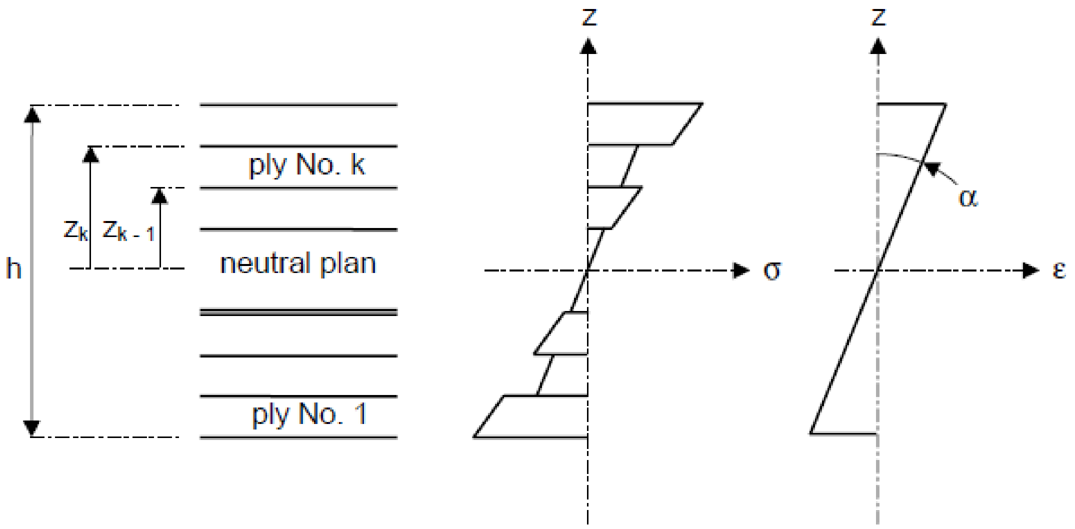


Figure 4.15: ply coordinates in the thickness direction

$$A^{*}_{ij} = \frac{A_{ij}}{t} = \frac{1}{t}\sum_{k=1}^{n} Q_{ij}\,(\theta^{\circ})(Z_k - Z_{k-1}) \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (4.20)$$

$$B_{ij} = \sum_{k=1}^{n} Q_{ij}\,(\theta^{\circ})\frac{1}{2}({Z^2}_k - {Z^2}_{k-1}) \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (4.21)$$

$$D_{ij} = \sum_{k=1}^{n} Q_{ij}\,(\theta^{\circ})\frac{1}{3}({Z^3}_k - {Z^3}_{k-1}) \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots (4.22)$$

$k \equiv$ Fiber coordinate system

A_{ij}: laminate stiffness matrix (membrane)

B_{ij}: laminate stiffness matrix (membrane/bending coupling)

D_{ij}: laminate stiffness matrix (bending)

General laminates subjected to plane stress and bending loads

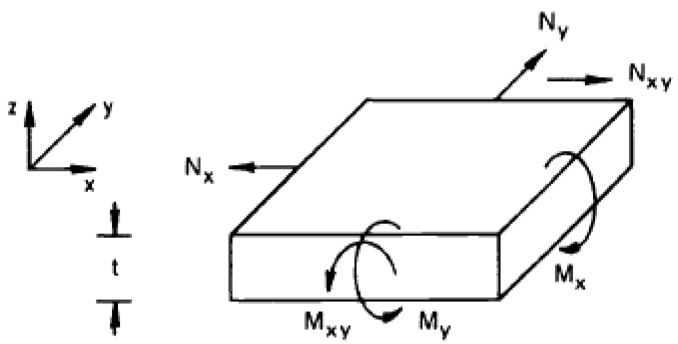


Figure 4.16: Stress and moments results for a laminate.

$$\begin{bmatrix} N_x \\ N_y \\ N_s \\ M_x \\ M_y \\ M_s \end{bmatrix} = \begin{bmatrix} A_{xx} & A_{xy} & A_{xs} & B_{xx} & B_{xy} & B_{xs} \\ A_{xy} & A_{yy} & A_{ys} & B_{xy} & B_{yy} & B_{ys} \\ A_{xs} & A_{ys} & A_{ss} & B_{xs} & B_{ys} & B_{ss} \\ B_{xx} & B_{xy} & B_{xs} & D_{xx} & D_{xy} & D_{xs} \\ B_{xy} & B_{yy} & B_{ys} & D_{xy} & D_{yy} & D_{ys} \\ B_{xs} & B_{ys} & B_{ss} & D_{xs} & D_{ys} & D_{ss} \end{bmatrix} \begin{bmatrix} \varepsilon_x \\ \varepsilon_y \\ \tau_s \\ \kappa_x \\ \kappa_y \\ \kappa_s \end{bmatrix} \dots\dots\dots. (4.23)$$

Bending of symmetric laminates

$$\begin{bmatrix} M_x \\ M_Y \\ M_S \end{bmatrix} = \begin{bmatrix} D_{xx} & D_{xy} & D_{xs} \\ D_{xy} & D_{yy} & D_{ys} \\ D_{xs} & D_{ys} & D_{ss} \end{bmatrix} \begin{bmatrix} \kappa_x \\ \kappa_y \\ \kappa_s \end{bmatrix} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.24)$$

Stress and moment resultants for a laminate

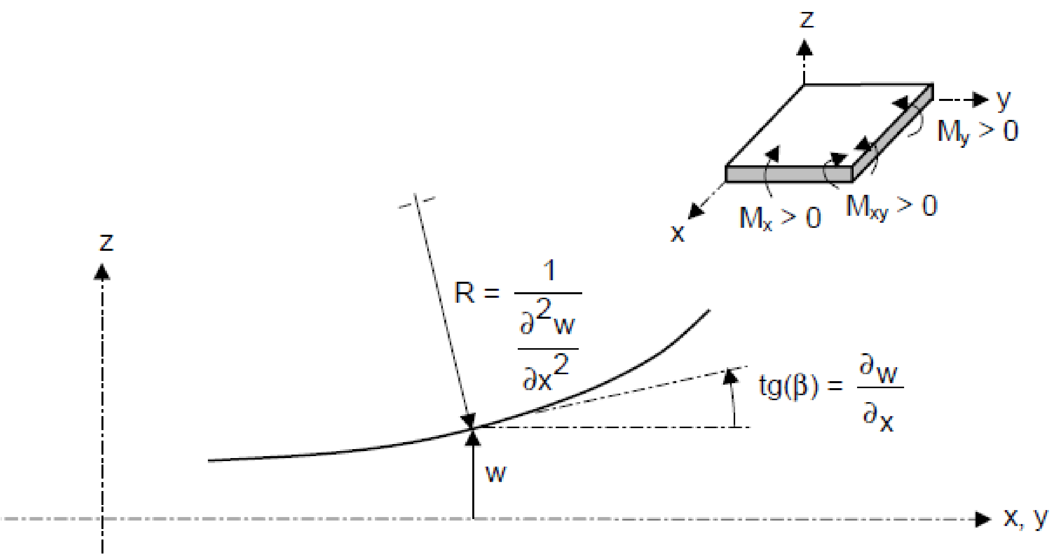


Figure 4.17: moment resultants for a laminate

R ≡ beam radius of curvature at a given point

β ≡ beam curvature at a given point

$$\kappa_x = -\frac{d^2w}{dx^2} = \frac{M}{EI} or\ {\partial^2 w_o}/{\partial x^2} \dots\dots\dots\dots\dots\dots\dots\dots\dots (4.25)$$

$$\kappa_y = -\frac{\partial^2 w}{\partial y^2} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.26)$$

$$\kappa_s = -\frac{\partial^2 w}{\partial x \partial y} \quad (4.27)$$

$\kappa_x, \kappa_y and \kappa_s$ The Kirchhoff shear forces.

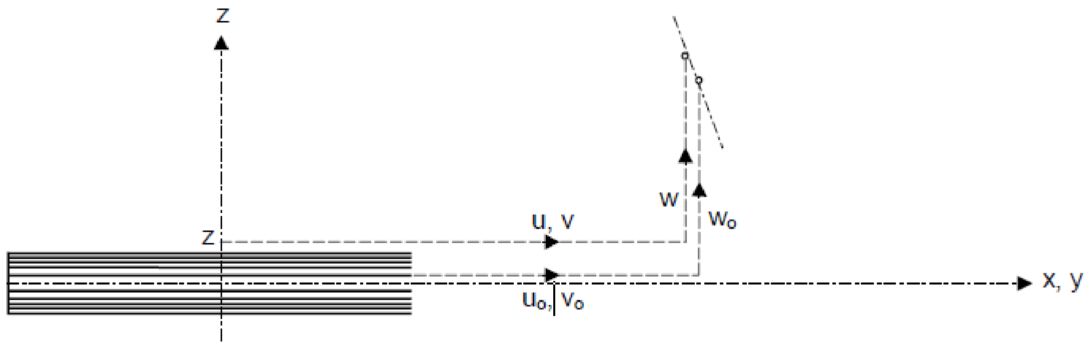


Figure 4.18: displacements from the neutral plane

$$u = u_0 - z\frac{\partial w_0}{\partial x} \quad (4.28)$$

$$v = v_0 - z\frac{\partial w_0}{\partial y} \quad (4.29)$$

$$w = w_0 \quad (4.30)$$

Where, $u_{0,}\ v_0\ and\ w_0$ represent displacements from the neutral plane in the coordinate system (x, y, and z).

We deduce (by deriving with respect to coordinates) the corresponding non-zero strains:

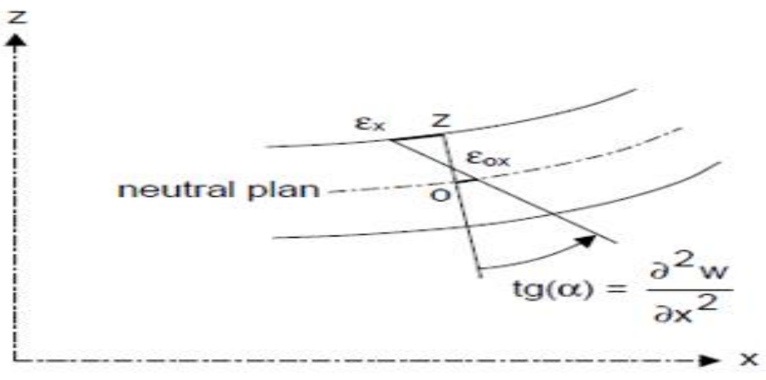


Figure 4.19: strains at any point at position z

Where ε_{0x} $\varepsilon_{oy}$and τ_s represent strains at a point located on the neutral plane andε_x , $\varepsilon_y$andγ_s represent strains at any point at position z.

$$\varepsilon_x = \varepsilon_{0x} - z\frac{\partial^2 w_0}{\partial x} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots . (4.31)$$

$$\varepsilon_y = \varepsilon_{oy} - z\frac{\partial^2 w_0}{\partial y} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.32)$$

$$\gamma_s = \gamma_s - 2z\frac{\partial^2 w_0}{\partial \mathrm{x}\partial \mathrm{y}} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.33)$$

Calculate the module from equation

$$\mathrm{E}_x = A_{xx} - \frac{{A_{xy}}^2}{A_{yy}} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots . (4.34)$$

$$\mathrm{E}_y = A_{yy} - \frac{{A_{xy}}^2}{A_{xx}} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.35)$$

$$\nu_{xy} = \frac{A_{xy}}{A_{yy}} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.36)$$

$$\nu_{yx} = \frac{A_{xy}}{A_{xx}} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.37)$$

$$G_{xy} = A_{sx} \dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots\dots (4.38)$$

Aileron load carrying structure sizing

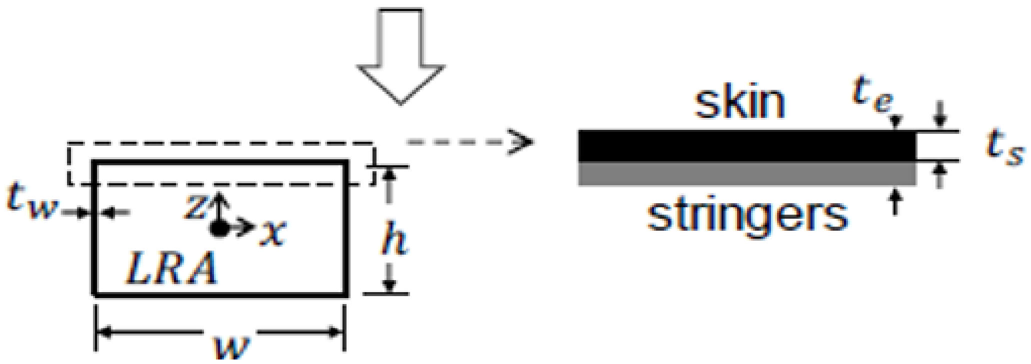


Figure 4.20: aileron box geometry

$$F_{mat,s} = 200MPa \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots..\ldots.. (4.39)$$

$F_{mat,s}$ ≡ material shear allowable for loads.

$$F_{mat,t} = \text{E}\varepsilon \Rightarrow \text{E} = 0.325\text{E}_{xo} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots.\ldots. (4.40)$$

$\varepsilon = 0.004$, $\text{E}_{xo} = 139GPa$

$F_{mat,t} = 0.325 * 139 * 0.004 * 10^6$=180.7MPa

$F_{mat,t}$ ≡ material tensile allowable for ultimate

$$P = \frac{SF.M_{LRA}}{h} = \frac{1.5*19.72267185}{0.060595554} = 488.2207644N \ldots\ldots\ldots\ldots (4.41)$$

SF ≡ Load factor

P ≡ is the load on the cover panel due to the bending moment.

$$F_b = A * \sqrt{\frac{P}{WL_r}} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots.. (4.42)$$

$$= 150 * \sqrt{\frac{488.2207644}{0.07985524 * 0.4323333334}} = 1783.767767 \ {}^{N}/_{m}$$

F_b ≡ buckling allowable

W ≡ Distance between mian spar and lading edg

l_r ≡ rib spacing(along the LRA)

$$Q = \frac{SF.T_{LRA}}{2Wh} = \frac{1.5 * 1.259}{2 * 0.07985524 * 0.060595554} = 195.14 \, {}^{N}/_{m} \ldots\ldots..\ldots (4.43)$$

Q ≡ stiffness matrix

T_LRA ≡ torsion load at the station

h ≡ length of spar

$$t_e = \left(\frac{\frac{P}{W}}{F_b}\right) = \left(\frac{\frac{488.2207644}{0.07985524}}{17837.67767}\right) = 0.34 \ldots\ldots\ldots\ldots.\ldots. (4.44)$$

t_e ≡ buckling and fatique the up and lower covers have different thickness

$$t_{s_{strength}} = \left(Max \left(\frac{\left(\frac{P}{W}\right)}{F_{mat,t}} \right), \left(\frac{\left(\frac{Q}{K_{S=1}}\right)}{F_{mat,S}} \right) \right) = \left(\frac{\left(\frac{488.2207644}{0.07985524}\right)}{180.7} \right) = 0.034 .. (4.45)$$

$$K_S = \frac{t_s}{\frac{t_s}{t_{stringer}}} \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots \dots .. (4.46)$$

K_s ≡ is skin ratio

t_S ≡ skin thickness =0.0300305

$$t_w = K_w K_{NO,w} * \frac{SF}{0.8F_{mat,S}} \left(\frac{|S_{LRA}|}{2h} + \frac{|T_{LRA}|}{2Wh} \right) = 0.000976 \dots \dots \dots (4.47)$$

t_w ≡ thickness spar web

$K_w K_{NO,w}$ ≡ the factor adesign varible (at each station)

S_{ALT} ≡ Shear load at the station

$$t_{rib,w} = K_{NO,rib} \frac{SF. ll_{aero}. l_r \cos \Lambda_{LRA}}{h. F_{mat,S}} + 0.003h = 0.1817 \dots \dots \dots \dots \dots \dots \dots . (4.48)$$

$t_{rib,w}$ ≡ thickness rib web

$$t_{rib,c} = 2K_{NO,rib} \frac{1}{hF_{mat,t}} . \frac{SF. ll_{aero}. l_r \cos \Lambda_{LRA} . W}{8} . \frac{1}{h} = 0.006176 \dots \dots \dots \dots (4.49)$$

$t_{rib,c}$ ≡ Thickness rib cap

Tsai and Wu postulated that a failure surface in six-dimension stress space exists in the form.

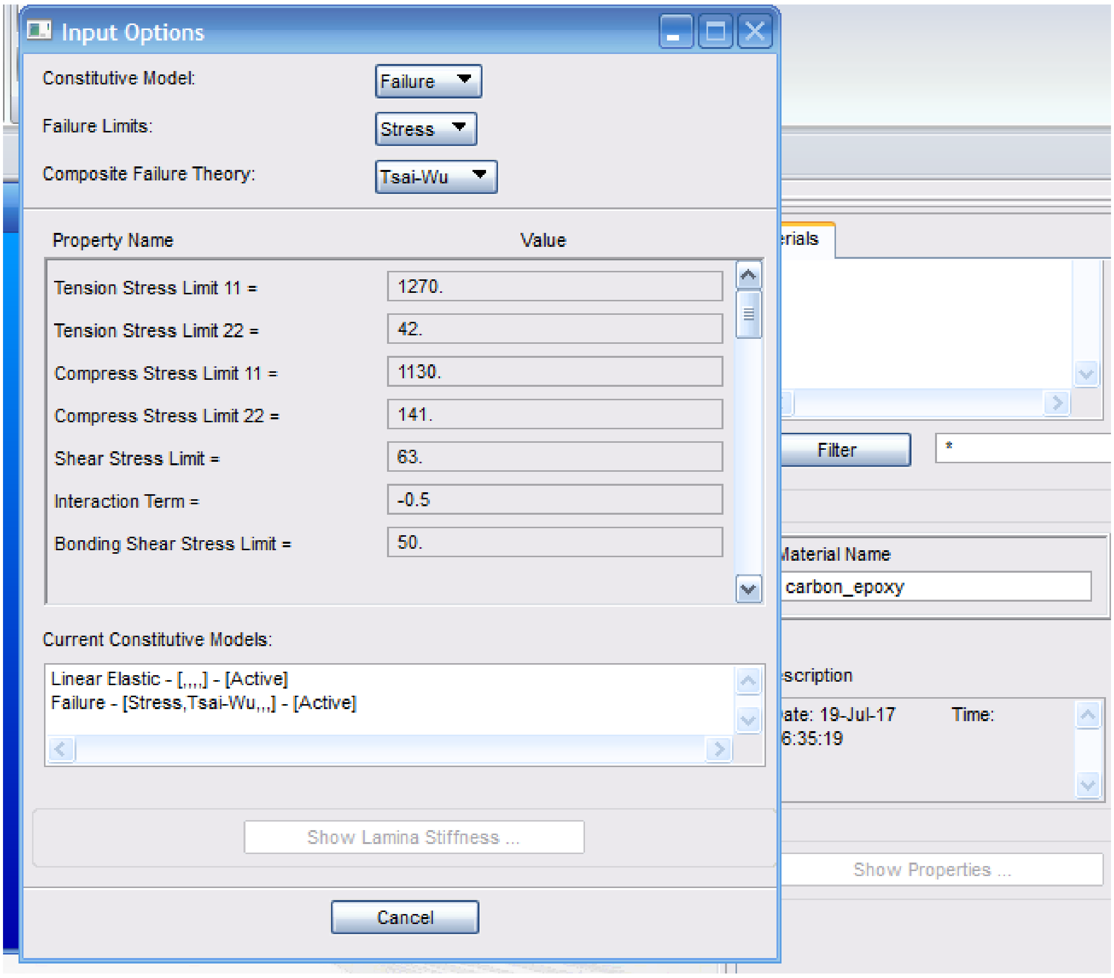


Figure 4.21: input failure

$F_i\sigma_i + F_{ij}\sigma_i\sigma_j = 1$

$$i, j = 1 \ldots, 6 \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots \ldots .. (4.50)$$

Where in F_i *and* F_{ij} are strength tensor.

Orthotropic lamina under plane stress conditions:

$$F_1\sigma_1 + F_2\sigma_2 + F_{11}\sigma^2{}_1 + F_{22}\sigma^2{}_2 + F_{66}\tau^2{}_{12} + 2F_{12}\sigma_1\sigma_2 < 1 \ldots \ldots \ldots (4.51)$$

The terms that are linear in the stresses are useful in representing different strengths in tension and compression. The terms that are quadratic in the stresses are the less usual terms to represent an ellipsoid in stress space. However, the independent parameter F_{12} is failure criterion on the dependent coefficient $2H = \frac{1}{X^2}$ in the Tsai-Hill in the 1-and 2-directions.

Under tensile load

$$F_1X_t + F_{11}X^2{}_t < 1 \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots \quad (4.52)$$

Under compressive load

$$F_1X_c + F_{11}X^2{}_c < 1 \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots \quad (4.53)$$

Upon simultaneous solution of Equations (4.52) and (4.53),

$$F_1 = \frac{1}{X_T} + \frac{1}{X_C} \qquad F_{11} = -\frac{1}{X_TX_C} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots.. \quad (4.54)$$

Similarly.

$$F_2 = \frac{1}{Y_T} + \frac{1}{Y_C} \qquad F_{22} = -\frac{1}{Y_TY_C} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots.\ldots\ldots\ldots\ldots.\ldots\ldots \quad (4.55)$$

Similar reasoning, along with our observation that the shear strength in principal material coordinates is independent of shear stress sign, leads to.

$$F_6 = 0 \qquad F_{66} = -\frac{1}{S^2{}_{12}} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots..\ldots\ldots.\ldots. \quad (4.56)$$

Note that for equal strengths in tension and compression ($X_t = -X_c$ and$Y_t = Y_c$).

The coefficient F_{12} requires biaxial testing. Let σ_{biax} be the equal biaxial tensile stress ($\sigma_1 = \sigma_2$ at failure. If it is known, then:

$$F_{12} = \frac{1}{2\sigma^2{}_{biax}}\left(1 - \left(\frac{1}{X_T} + \frac{1}{X_C} + \frac{1}{Y_T} + \frac{1}{Y_C}\right)\sigma_{biax} + \left(\frac{1}{X_TX_C} + \frac{1}{Y_TY_C}\right)\sigma^2{}_{biax}\right) \ldots.. \quad (4.57)$$

Otherwise,

$$F_{12} = f^1\sqrt{F_{11}F_{22}} \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots.. \quad (4.58)$$

Where-1.0$\leq f^1 \leq 1.0$. the default value of $f^1$is zero.

Manufacturing sample for composite material (carbon fiber unidirectional)

Figure 4.22: manufacturing sample for composite material

5 Chapter Five: Results and Discussion

5.1 Results

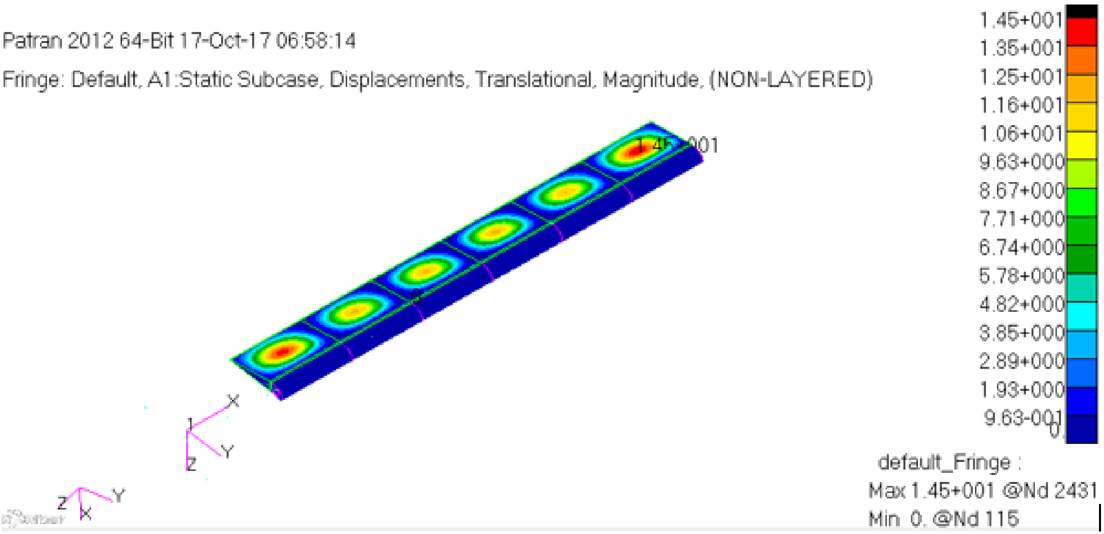


Figure 5.1: displacement results

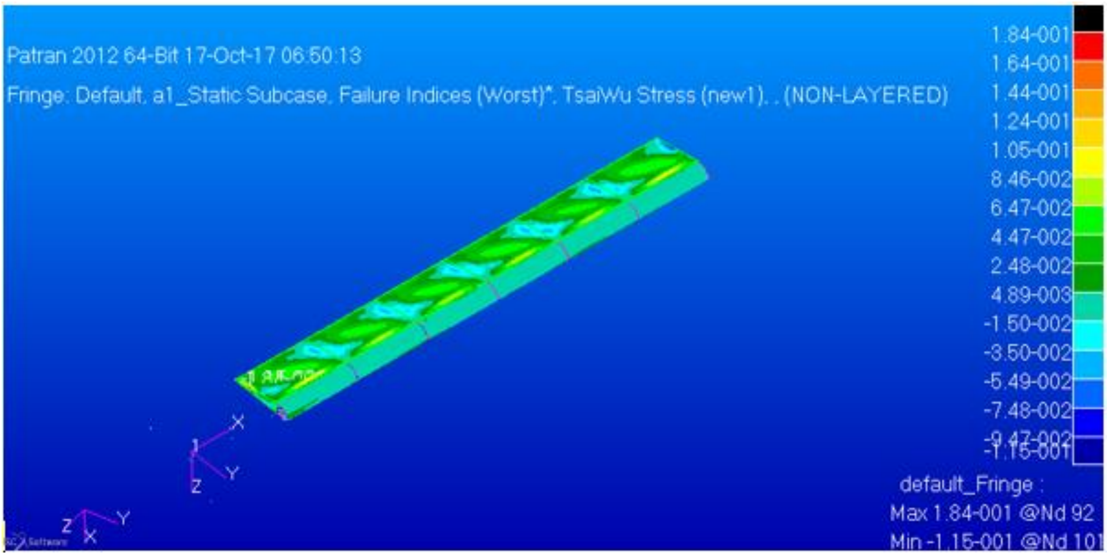


Figure 5.2: Failure index by Tsai-Wu.

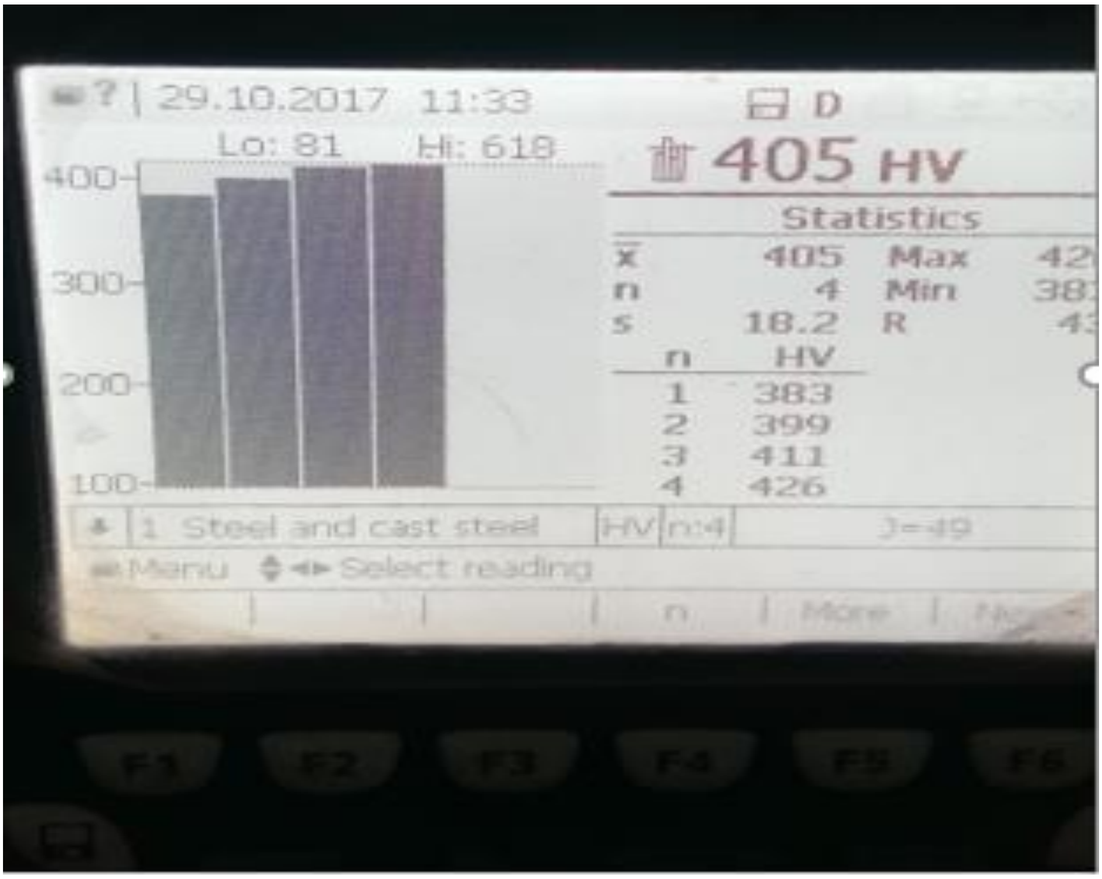


Figure 5.3: hardness test results

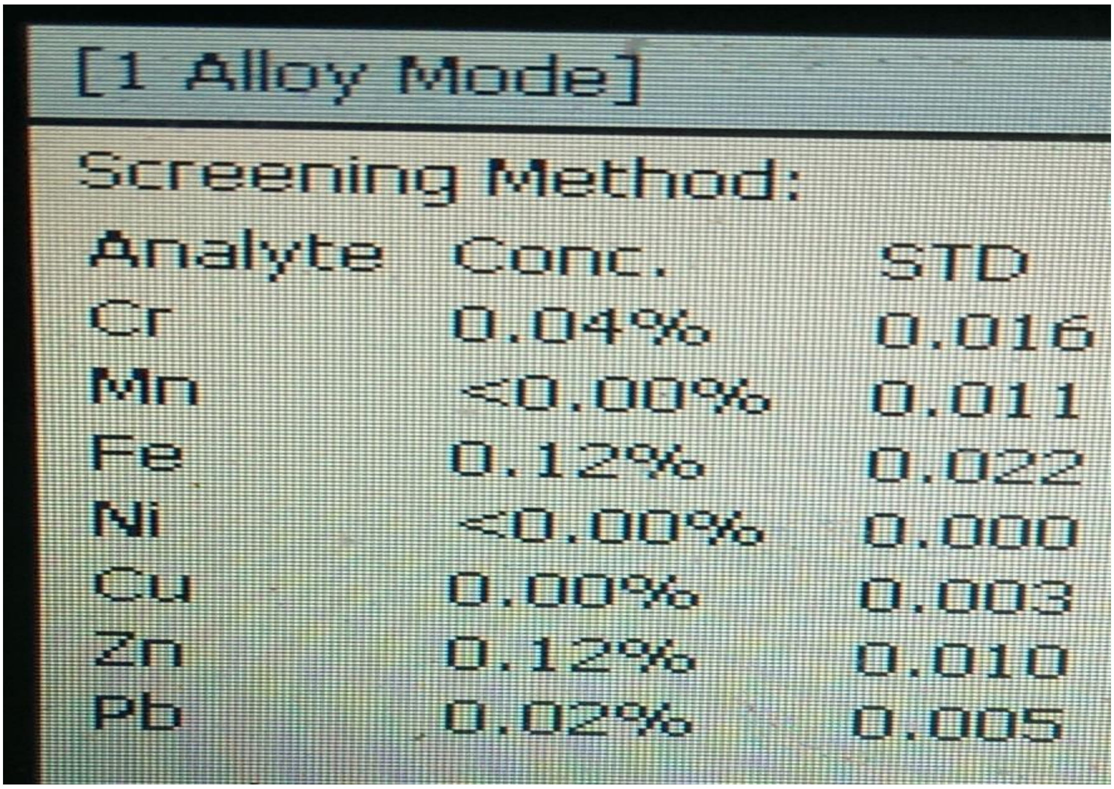


Figure 5.4: positive material analyzing test results

Figure 5.5: impact test results

5.2 Discussion

In [figure 5.1]. It can see that a highest displacement at outer sections of the aileron due to aileron fixed at medal sections and minimum at fixed sections.

[Figure 5.2] shows the maximum and minimum failure index, if the failure index lower than one; the design is okay. And if it's higher than one, there will be over design condition. When we calculate the skin, rib and spar thickness we found that the value of spar thickness is very small, due to this result the spar is neglected from aileron part.

[Figure 5.3] is showing the result of hardiness test which applied to composite material carbon fiber (carbon/epoxy). The value of hardness is evaluated by taken an average of four values in different position, the maximum value is 426 HV, the minimum value is 383 HV (hardness vicar) and the average value is 405 HV.

[Figure 5.4] is showing the result of positive material analyzing (PMA) test which describe the material element component, the maximum element percentage (carbon) is 70% and the minimum value percentage (Pb) is 2%.

[Figure 5.5] display result of impact test applied over two samples of composite material carbon fiber (carbon/epoxy). The first sample absorbed impact value of 60.5 Joules and the second sample absorbed 60.9 Joules.

6 Chapter Six: Conclusion and Recommendation

6.1 Conclusion

We used the composite material (carbon/epoxy) in manufacturing of aircraft aileron structure.

Simulation has done by use software program (CATIA, PATRAN and NASTRAN) to represent the deformation of the model, the deformation result is very small and failure index result is within a range.

The structure sample has manufactured and tested, the resultant test is 60.5 joules for impact test, 70% of carbon as a result for the positive material analyzing (PMA) and the hardness test result is 405 HV.

6.2 Recommendation

As we saw that the results of tests were good and acceptable, we recommend to students who want to continue in this project to complete the other testing such as bending test and compressive test and compare the resultant test with simulation.

6.3 Future Work

1. Availability of high performance resins meeting production.
2. Availability of relevant report and environment data.
3. Fabricate the whole airframe of aircraft from composite materials.

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