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Structural Weight Optimization of Aircraft Wing Component Using FEM Approach.

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Abstract—
One of the main challenges for the civil aviation industry is the reduction of its environmental impact by better fuel efficiency by virtue of Structural optimization. Over the past years, improvements in performance and fuel efficiency have been achieved by simplifying the design of the structural components and usage of composite materials to reduce the overall weight of the structure. This paper deals with the weight optimization of transport aircraft with low wing configuration. The Linear static and Normal Mode analysis were carried out using MSc Nastran & Msc Patran under different pressure conditions and the results were verified with the help of classical approach. The Stress and displacement results were found and verified and hence arrived to the conclusion about the optimization of the wing structure.
Keywords: Transport Aircraft, Composite materials, Wing Optimization, Stress Analysis, MSc Nastran & Msc Patran.

I. INTRODUCTION

The wings are airfoil attached to each side of the fuselage and are the main lifting surfaces that support the airplane in flight. There are numerous wing design, size and shape were used by the various manufacturers. Each fulfils a certain need with respect to the expected performance for the particular airplane. Wings may be attached at the top, middle, or lower portion of the fuselage. These designs are referred to as high wing, mid wing and low wing configurations respectively. The number of wings can also vary, an Airplane with a single set of wing is referred as a monoplane, while those with two sets called biplane.

Many high-wing airplanes have external braces or wing struts which transmit the flight and landing loads through the struts to the main fuselage structure. Since the wing struts are usually attached approximately halfway out on the wing, this type of wing structure is called semi-cantilever. A few high-wing and most low-wing airplanes have a full cantilever wing designed to carry the loads without external struts.

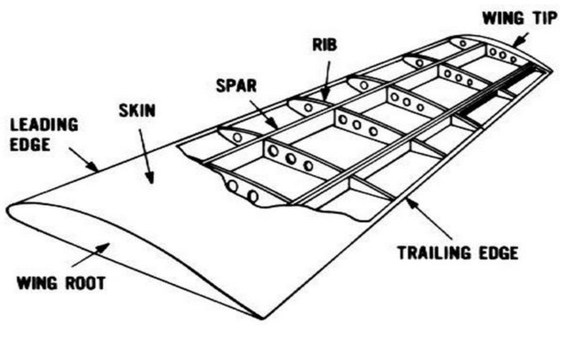


Figure 1: Wing Structure

II. AIR FOIL SELECTION

This section is devoted to the process to determine airfoil section for a wing. It is appropriate to claim that the airfoil section is the second most important wing parameter; after wing planform area. The airfoil section is responsible for the generation of the optimum pressure distribution on the top and bottom surfaces of the wing such that the required lift is created with the lowest aerodynamic cost (i.e. drag and pitching moment). Although every aircraft designer has some basic knowledge of aerodynamics and airfoils; but to have a uniform starting point, the concept of airfoil and its governing equations will be reviewed. The section begins with a discussion on airfoil selection or airfoil design. Then basics of airfoil, airfoil parameters and most important factor on airfoil section will be presented. A review of NACA4 - the predecessor of the present NASA5-airfoils will be presented later, since the focus in this section is on the airfoil selection. The criteria for airfoil selection will be introduced and finally the procedure to select the best airfoil is introduced. The section will be ended with a fully solved example to select an airfoil for a candidate wing.

III. AIR FOIL DESIGN

The primary function of the wing is to generate lift force; this will be generated by a special wing cross section called air foil. Wing is a three dimensional component, while the airfoil is two dimensional section. Because of the airfoil section, two other outputs of the airfoil and consequently the wing are drag and pitching moment. The wing may have a constant or a non-constant cross-section across the wing. There are two ways to determine the wing air foil section:

1. Airfoil design
2. Airfoil selection

The design of the airfoil is a complex and time consuming process and needs expertise in fundamentals of aerodynamics at graduate level. Since the airfoil needs to be verified by testing it in a wind tunnel, it is expensive too. Large aircraft production companies such as Boeing and Airbus have sufficient human experts (aerodynamicists) and budget to design their own airfoil for every aircraft but small aircraft companies, experimental aircraft producers and homebuilt manufacturers do not afford to design their airfoils. Instead they select the best airfoil among the current available airfoils that are found in several books or websites.

IV. WING CALCULATION

1) LIFT COEFFICIENT

CLmax = 2×W/ ρ× (VSTALL)2×S

Where,
CLmax =Coefficient of lift
W =Maximum takeoff weight(Kg)
VSTALL =Cruise speed x0.25(Km/h)
S =Wing area(m^2)
ρ =Density(Kg/m^3)

2) CHORD LENGTH

C=S/b

Where,
C = Chord length(m)
S = Wing area(m^2)
b = Wing span(m)

3) INDUCED DRAG CO EFFICIENT

CDi = CL2 / (π×e×AR)

Where,
e = Oswald efficiency factor
AR = Aspect Ratio

4) OSWALD EFFICIENCY FACTOR

e = 4.61(1-0.045AR0.68) (cos(Λ LE))0.15-3.1

Where,
ΛLE =sweep back angle

5) PROFILE DRAG CO EFFICIENT

CDo = Cfe× (Swef / Sref)

Where,
Cfe = Equivalent skin friction co efficient
Swef / Sref =Wetted area ratios

6) TOTAL DRAG CO EFFICIENT

CD = CDo + CDi

Where,
CDi = Induced drag co efficient
CDo = Profile drag co efficient

7) ROOT CHORD

Croot = 2×S / b*(1+ƛ)

Where,
ƛ = Taper ratio

8) TIP CHORD

Ctip = ƛ×Croot

Where,
Ctip =Tip chord
Croot = Root chord

9) EXPOSED WING SPAN

Semi span =b/2

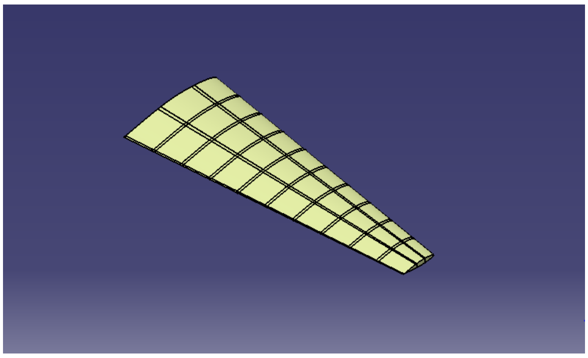
Table 1: Wing Design Parameters

SL.NO	Wing Design Parameters	Value
1	Lift co-efficient	1.32
2	Chord length	4.28 m
3	Induced Drag Co-efficient	0.106
4	Oswald Efficiency Factor	0.65
5	Profile Drag Co Efficient	0.016
6	Total Drag Co Efficient	0.122
7	Root Chord	6.436 m
8	Tip Chord	2.117 m
9	Exposed Wing Span	17.15 m
10	Swept back angle	20°

V. DESIGN METHODOLOGY

A swept back wing is a type of aircraft wing configuration where, the root of the wing is lower than the wing tip, when the aircraft is seen from the front along the horizontal axis. Dihedral contributes positively to the stability of the aircraft along the roll-axis (spiral mode). As the aircraft bank, the swept back wing tends to roll the aircraft and helps to restore indirectly the wing level.

To develop 3D model Catia software is used CATIA (Computer Aided three Dimensional application) which is design software developed by Dassaults systems to meet the complicated design requirements in the field of Aerospace and Automotive. Here we design swept back wing and legs separately in CATIA. CATIA was developed for Boeing later due to its advantage many other organizations used.



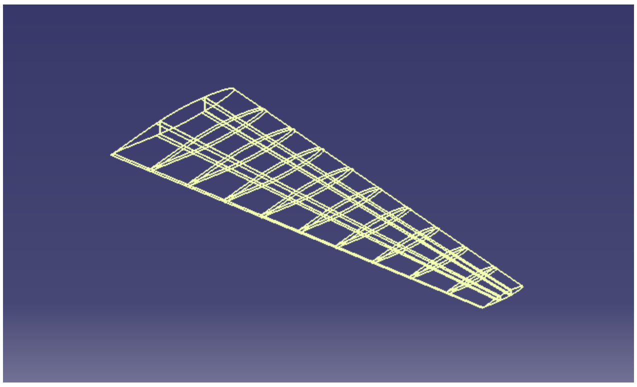


Figure 2: 3D model of Wing Structure

VI. MATERIALS USED FOR THE CONSTRUCTION OF SWEPT BACK WING

A. Al ALLOY

Alloys composed of mostly of Aluminum have been very important in aerospace manufacturing since the introduction of metals in the aircraft. Aluminum alloys are light weight and has high strength. Aluminum alloys have good thermal strength. The most commonly used Aluminum alloys for airframe construction are precipitation hardening alloys in 2XXX and 7XXX series.

B. 2XXX SERIES Al ALLOYS

This series Al alloys contains copper, manganese, magnesium and zinc. These alloys have low crack growth rates and better fatigue performance. These are used on the lower wings and body skin of the aircraft.

C. 7XXX Al ALLOYS

This alloy consists of aluminum, zinc and magnesium. Copper often added to improve stress corrosion cracking resistance. They are used in light weight military and civil aircrafts.

Table 2: Aluminum 2024 T3 Mechanical Property

Sl. NO	Properties	value	units
1	Density	2.78E-6	**[kg/mm3]**
2	Young's Modulus,	73.1	**[GPa]**
3	Shear Modulus,	28	**[GPa]**
4	Poison's ratio	0.33	
5	Ultimate strength	483	**[MPa]**
6	Yield strength	385	**[MPa]**
7	Shear strength	283	**[MPa]**

D. *ADVANTAGES OF ALUMINIUM*

1. *Aluminum combined with an appropriate alloy ensures steel durability.*
2. *It may be easily formed in the course of all machining processes, such as rolling, embossing, forging and die-casting.*
3. *Aluminum structures have considerable insulation properties securing from air and light activity*
4. *Such structures are light, which facilitates assembly and transportation.*
5. *Aluminum has a natural anti-corrosion layer which efficiently protects from environmental influences.*
6. *Aluminum requires little energy required in the processing process. Recycling saves 95% of the energy.*
7. *Aluminum as a resource is 100% recyclable.*

CRITERIA FOR CHOOSING CFRP MATERIAL:

The proposed aircraft wing structure is an innovative concept. The whole wing is fabricated with CARBON FIBRE REINFORCED PLASTICS (CFRP). The main advantages in this new design are very good integration, faster fabrication and assembly, weight reduction, possibility of thickness variation, less waste of raw material, higher passenger comfort level, longer structural life (less sensitive to fatigue), possibility of larger windows. There are also some disadvantages, although there are some possible solutions to overcome these disadvantages. The main disadvantages are electromagnetic interference, return of electric current, lightening protection, higher machinery investments and higher certification costs.

Table 2: Mechanical Property Carbon Fibre Reinforced Plastics (Cfrp).

Sl.No	Mechanical Properties	Value	Unit
1	Young's Modulus,E11	138.6	**GPa**
2	Young's Modulus,E22	82.7	**GPa**
3	Poisson's Ratio	0.25	**-**
4	Ultimate Strength	1450	**MPa**
5	Shear modulus, 12	4120	**MPa**
6	Shear modulus, 23	0.6	**MPa**
7	Shear modulus, 13	0.6	**MPa**
8	Yield Strength	600	**Mpa**
9	Density	1.76E-6	**Kg/mm^3**
10	Specific Gravity	1.6	**-**

VII. FINITE ELEMENT MODELLING OF WING COMPONENT

About FEA and Applications

THE finite element method is a numerical method that can be used for the accurate solution of complex engineering problems. The method was first developed in 1956 for the analysis of aircraft structural problems. Thereafter, within a decade, the potentialities of the method for the solution of different types of applied science and engineering problems were recognized. Over the years, the finite element technique has been so well established that today it is considered to be one of the best methods for solving a wide variety of practical problems efficiently. In fact, the method has become one of the active research areas for applied mathematicians. One of the main reasons for the popularity of the method in different fields of engineering is that once a general computer program is written, it can be used for the solution of any problem simply by changing the input data. A variety of specializations under the umbrella of the mechanical engineering discipline (such as aeronautical, biomechanical, and automotive industries) commonly use integrated FEM in design and development of their products. Several modern FEM packages include specific components such as thermal, electromagnetic, fluid, and structural working environments. In a structural simulation, FEM helps tremendously in producing stiffness and strength visualizations and also in minimizing weight, materials, and costs.FEM allows detailed visualization of where structures bend or twist, and indicates the distribution of stresses and displacements. FEM software provides a wide range of simulation options for controlling the complexity of both modeling and analysis of a system. Similarly, the desired level of accuracy required and associated computational time requirements can be managed simultaneously to address most engineering applications. FEM allows entire designs to be constructed, refined, and optimized before the design is manufactured. This powerful design tool has significantly improved both the standard of engineering designs and the methodology of the design process in many industrial applications. The introduction of FEM has substantially decreased the time to take products from concept to the production line. It is primarily through improved initial prototype designs using FEM that testing and development have been accelerated. In summary, benefits of FEM include increased accuracy, enhanced design and better insight into critical design parameters, virtual prototyping, fewer hardware prototypes, a faster and less expensive design cycle, increased productivity, and increased revenue. FEA has also been proposed to use in stochastic modeling for numerically solving probability models

VIII. PROCEDURE INVOLVED IN STRESS ANALYSIS

General procedure of FEM

i. Importing model or creating the model
ii. Discretization of the structure
iii. Applying material
iv. Applying load and boundary conditions
v. Analysis
vi. Solver
vii. Result

IX. STATIC ANALYSIS OF SWEPT BACK WING

Discretization of the Structure

The next step in the finite element method is to divide the structure region into subdivisions or elements. Hence, the structure is to be modeled with suitable finite elements. The number, type, size, and arrangement of the elements are to be decided. For Wing meshing is done by 1D and 2D elements.1D elements are used in spar flange section. Spar web is meshed manually using quad elements. Wing skin is meshed automatically by using Quad4 elements. Ribs are meshed by manually using quad and tri elements.1D elements are applied to the after meshing, equivalence is applied to delete duplicate node. Next is verification of mesh elements. First verified for element boundaries, duplicate elements, normals and geometry fit meshing Quad element is checked for error. Errors in quad element are

- Aspect ratio
- Warp
- Taper
- Skew

Table: 5 Type of Elements Used

Sl. no	***Component***	***Element Type***	***Number of elements***
1	*Spar Flange*	*Bar*	*360*
2	*Skin*	*Quad & Tria*	*5768*
3	*Spar Web*	*Quad*	*180*
4	*Ribs*	*Quad & Tria*	*348*

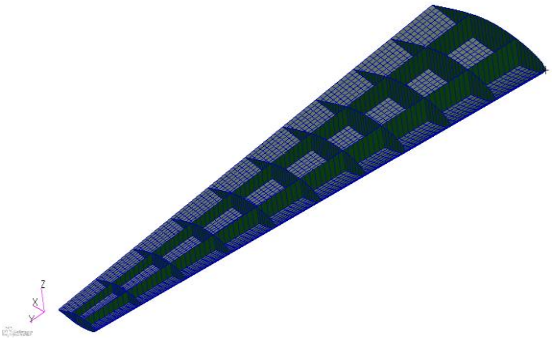


Figure 3: Finite Element Model

X. Properties

Sl. no	*Component*	*Element Type*	*properties*
1	*Spar Flange*	*Bar*	*Width=40 mm Height=12mm Material: Aluminum 2024 T3*
2	*Skin*	*Quad & Tria*	*Thickness=2mm Material: Aluminum 2024 T3*
3	*Spar Web*	*Quad*	*Thickness=15mm Material: Aluminum 2024 T3*
4	*Ribs*	*Quad & Tria*	*Thickness=3mm Material: Aluminum 2024 T3*

XI. Load calculation:

Limit Load = Take of weight x Lift
Lift Load = 4
Design load = Limit Load x Factor of Safety
Factor of Safety= 1.5
Load on semi span = Design load /2
Exposed wing area = (b/2)*(Croot+Ctip)
Pressure Load on wing = Load on semi span / wing area

XII. Analysis of Swept Back Wing Model

The Analysis results shown in below figures and the material wise results are tabulated. The validation of wing structure is done by Reserve Factor. The static analysis results are done by failure theories for both metallic and composite material.

XIII. Results of Aluminum 2024 T3

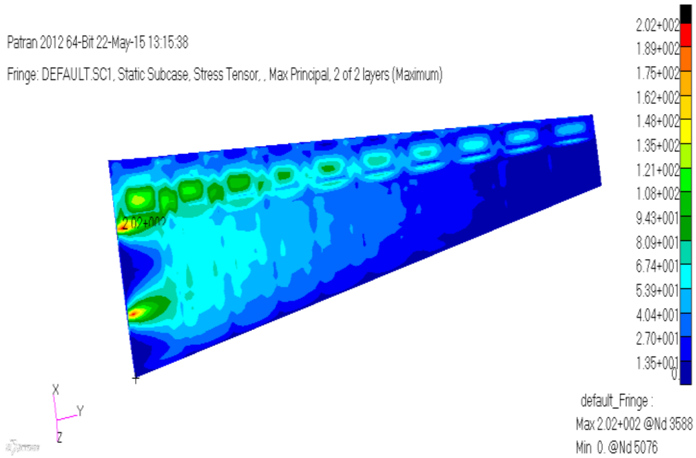


Figure 4: Max. Principal Stress = 202 N/mm^2

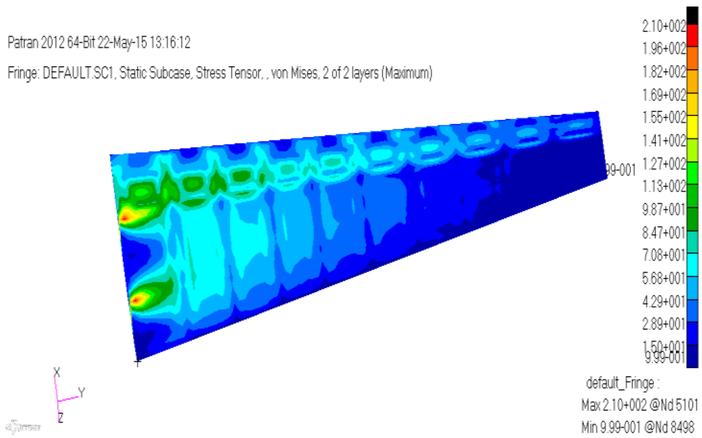


Figure 5: Von Mises Stress = 210 N/mm^2

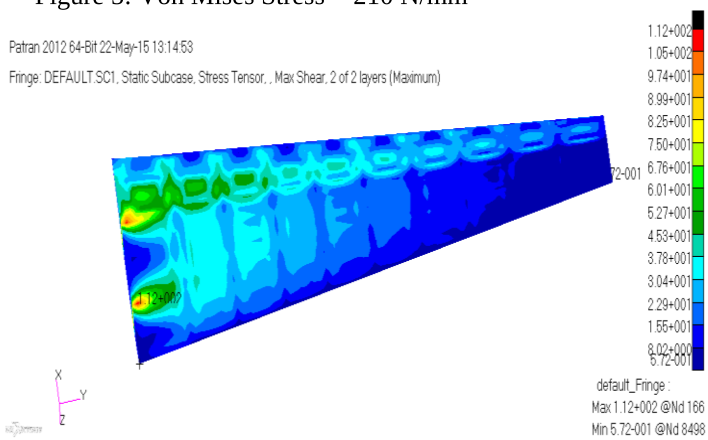


Figure 6: Max. Shear Stress = 112 N/mm^2

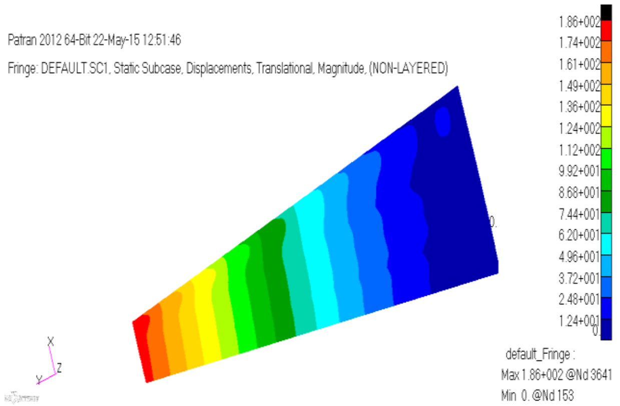


Figure 7: Max. Displacement = 186 mm

XIV. Results of CFRP

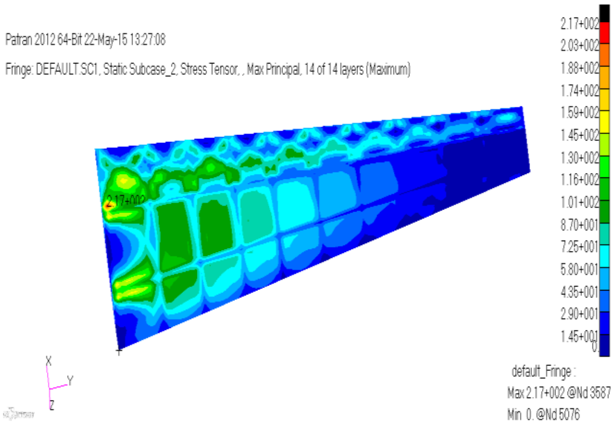


Figure 8: Max. Principal Stress = 217 N/mm^2

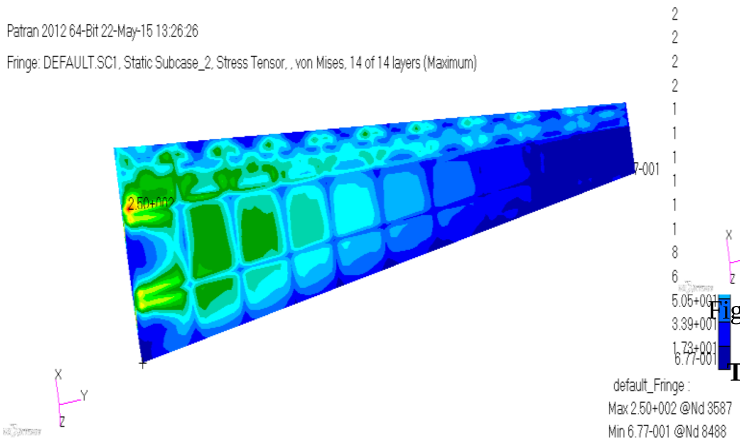


Figure 9: Von Mises Stress = 250 N/mm^2

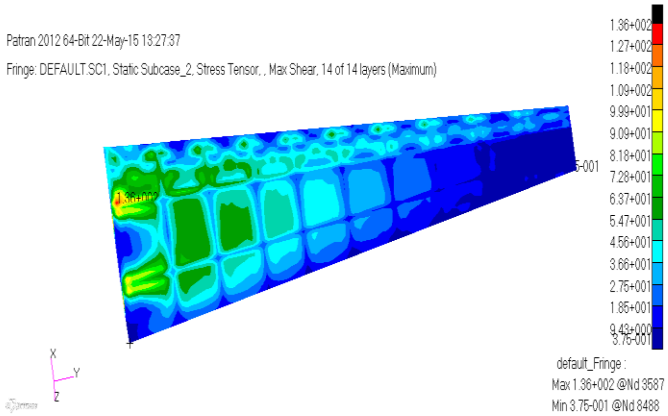


Figure 10: Max. Shear Stress = 136 N/mm^2

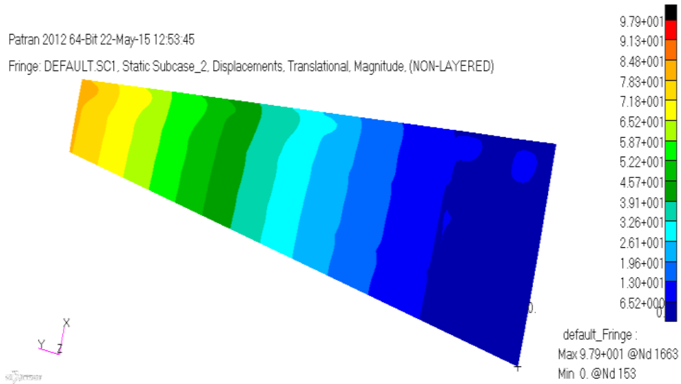


Figure 11: Max. Displacement = 97.9 mm

XIV. Failure Indices

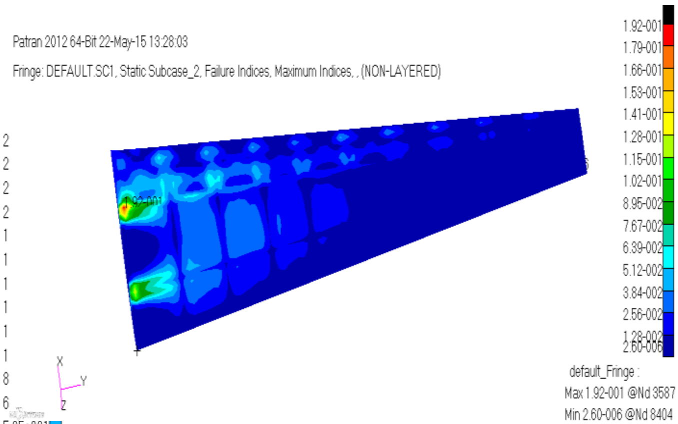


Figure 12: Failure Indices for CFRP = 0.19

Table: 7 Reserve Factor Table for aluminum 2024 T3

Sl.No	Type Of Stress	Allowable Stress	Obtained Stress	R.F
1	Max. Principal	385	202	1.90
2	Von Mises	385	210	1.83
3	Max. Shear	231	112	2.05

Table: 8 Reserve Factor Table for CFRP

Sl.No	Type Of Stress	Allowable Stress	Obtained Stress	R.F
1	Max. Principal	600	217	2.7
2	Von Mises	600	250	2.4
3	Max. Shear	360	136	2.6

XV. Normal Mode Analysis:

For Aluminum 2024 T3

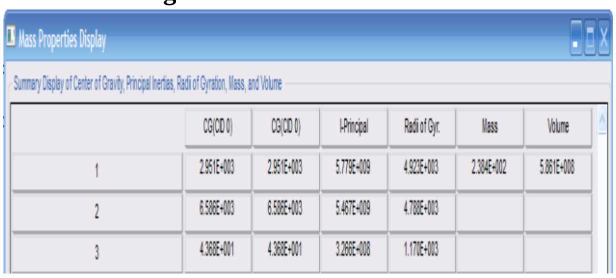
Sl. NO	Mode	Eigen Value	Frequency (Hz)
1	1st Bending	2.64	1.62
2	1st Twisting	14.7	3.84
3	2nd Bending	15.4	3.92
4	1st Axial	20.6	4.54
5	2nd Axial	21.4	4.63
6	2nd Twisting	23.3	4.82

For CFRP

Sl. NO	Mode	Eigen Value	Frequency (Hz)
1	1st Bending	3.35	1.83
2	1st Twisting	12.0	3.47
3	2nd Bending	13.7	3.71
4	1st Axial	16.6	4.08
5	2nd Axial	18.0	4.24
6	2nd Twisting	19.9	4.46

XVI. Structural Weight Optimization:

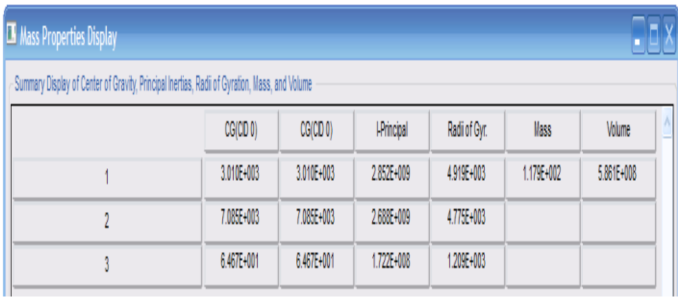
From the above analysis we have proved that the CFRP material is having high Strength to weight ratio. The initial **Aluminum 2024 T3 material mass is 238.6 Kg.**


Mass Properties Display

Summary Display of Center of Gravity, Principal Inertias, Radii of Gyration, Mass, and Volume

	CG(CID 0)	CG(CID 0)	I-Principal	Radii of Gyr.	Mass	Volume
1	2.951E+003	2.951E+003	5.779E+009	4.923E+003	2.384E+002	5.861E+008
2	6.586E+003	6.586E+003	5.467E+009	4.788E+003		
3	4.368E+001	4.368E+001	3.266E+008	1.170E+003		

Further modification of **CFRP material is having mass of 117.9 Kg** which is having **50.58%** less weight than the initial mass.


Mass Properties Display

Summary Display of Center of Gravity, Principal Inertias, Radii of Gyration, Mass, and Volume

	CG(CID 0)	CG(CID 0)	I-Principal	Radii of Gyr.	Mass	Volume
1	3.010E+003	3.010E+003	2.852E+009	4.919E+003	1.179E+002	5.861E+008
2	7.085E+003	7.085E+003	2.688E+009	4.775E+003		
3	6.467E+001	6.467E+001	1.722E+008	1.209E+003		

XVII. Conclusion:

The wing structure with internal components is designed and FE modeling is carried out for the above mentioned loads and Boundary conditions. The results for the Static Analysis is obtained and plotted and Reserve factor values are calculated.

The reserve factors for all the structural components are studied and scrutinized, the model is structurally good and can withstand the loads acting on it. It was also proved that the composite material is having high strength and is structurally stable for the loads applied. The structural behavior of various composite materials was also understood and obtained good knowledge on the practical stress analysis of composite structures using FEA process.

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