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STATIC & DYNAMIC ANALYSIS OF A TYPICAL AIRCRAFT WING STRUCTURE USING MSC NASTRAN

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Abstract:

The paper is about preliminary sizing and analysis of a trainer aircraft wing. The main objective is to fix an appropriate structure within the given envelope and to estimate the Gross take-off weight, wing loading, Stress distribution, low frequency vibrational modes, take-off distance and stall velocity. Sizing is done by using classical engineering theories and FEA packages. Skin and web are considered as shell elements. Flange, spar and stringer are considered as beam elements. From the analysis structure has been optimally designed which satisfies the strength and stability criteria. The detailed design of trainer aircraft wing structure is modelled using CATIA V5 R20. Then stress analysis of the wing structure is carried out by using the finite element approach with the help of MSC NASTRAN/PATRAN to find out the safety factor of the structure.

1. INTRODUCTION

A wing is a surface used to produce an aerodynamic force normal to the direction of motion by traveling in air or another gaseous medium, facilitating flight. It is a specific form of aerofoil. The optimum structural design of an A/C wing is an important factor in the performance of the airplanes i.e. obtaining a wing with a high stiffness/weight ratio and sustaining the unexpected loading such as gust and manoeuvring situations. This is accomplished by studying the different design parameters required to specify the wing geometry. The idea of the structural optimization in the classical sense, has been considered to be the minimization of structural mass by varying member sizes or shell thickness of a model in which the geometry remains unchanged therefore many studies have been made during the last years to find the structural optimization. In an aircraft wing structure ribs and spars are provided to support and give rigidity to the wing section. Although the major focus of structural design in the early development of aircraft was on strength, now structural designers also deal with fail-safety, fatigue, corrosion, maintenance and inspect ability, and predictability. Modern aircraft structures are designed using a semi-monologue concept. A basic load-carrying shell reinforced by Frames and longerons in the bodies, and a skin stringer construction supported by spars and ribs in the surfaces. Proper stress levels, a very complex problem in highly redundant structures, are calculated using versatile computer matrix methods to solve for detailed internal loads. Modern finite element models of aircraft components include tens-of-thousands of degrees-of-freedom and are used to determine the required skin thicknesses to avoid excessive stress levels, deflections, strains, or buckling. The goals of detailed design are to reduce or eliminate stress concentrations, residual stresses, fretting corrosion, hidden undetectable cracks, or single failure causing component failure.

2. PROBLEM STATEMENT

The wing structure experience various types of loads during each phase of the flight which includes take-off, climb, cruise, loiter, landing, touch-down. In each segment there is variation in load factor which induces various types of stresses in various components of aircraft body. This problem can be simplified by considering wing as a cantilever beam whose one end is fixed in the fuselage and the another end is free. The loading condition on a wing is equivalent to the uniform varying load throughout the wing. The various types of stresses and its intensity induced in the skin of the aircraft during take-off and the first six modes of vibration which are possible when the aircraft in at ground will be explained in this project. As all the details required for solving this problem are not available appropriate assumptions are made wherever required to simplify the problem. To simplify the problem the assumptions made are as follows

- The aerofoil used in Aircraft wing is supercritical aerofoil NACA 64A215 at root and NACA 64A210 at tip.
- The material used for the whole structural and modal analysis purpose is aluminium alloy with density of 2700 kg/m^3, young modulus of 68300 Mpa, and poison's ratio of 0.34.
- The boundary conditions applied to the FEA model is that the root section of the aerofoil as well as the spars is fixed so that the degree of freedom is restricted in all the six directions.
- The loading condition is found using the maximum take-off weight and maximum climb angle which is allowed for this aircraft from any airport.
- Material is homogenous.
- Material is isotropic.
- Material is elastic
- The stringers carry only axial stresses.
- The skins carry only shearing stresses.
- The spar carry bending load.

3. GEOMETRICAL CONFIGURATION

The Wing is designed in CATIA V5 R20. The wing consist of 16 ribs in transverse direction and 2 spars in longitudinal direction. AA 2024-T351 is used in current wing structure due to high strength and fatigue resistance properties. The ultimate tensile strength of this material is 427 Mpa and yield strength is 324 Mpa.

TABLE 1. GEOMETRIC PARAMETERS OF THE DESIGN

Root Chord	2400 mm
Tip Chord	700 mm
Semi span length	5500 mm
Exposed span length	4750 mm
Aerofoil (root)	NACA 64A215
Aerofoil (tip)	NACA 64A210
Aircraft Weight	14000 N

Load Factor	6g
Design Factor	1.5
Front spar	18 to 25 % of chord
Rear spar	62 to 70 % of chord
Surface area	8.525 m^2

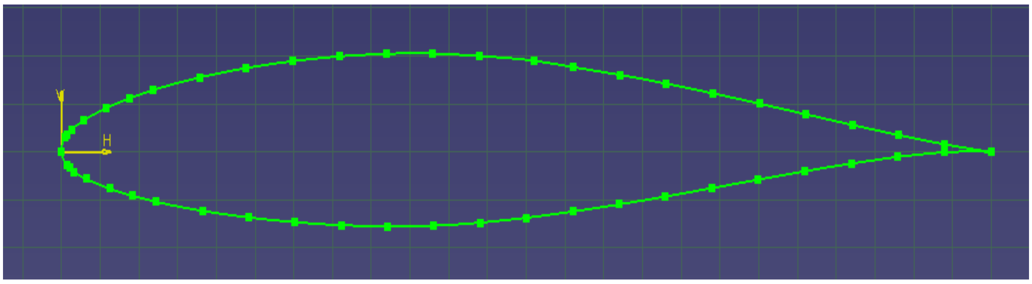

Figure 1. Aerofoil (Root)

Above Figure 1 shows the NACA 64A215 Aerofoil of chord 2400 mm used in root section.

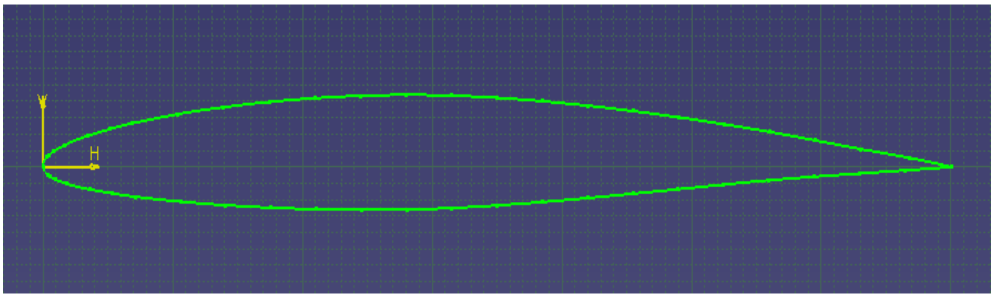

Figure 2. Aerofoil (Tip)

Above Figure 2 shows the NACA 64A210 Aerofoil of chord 700 mm used in tip section.

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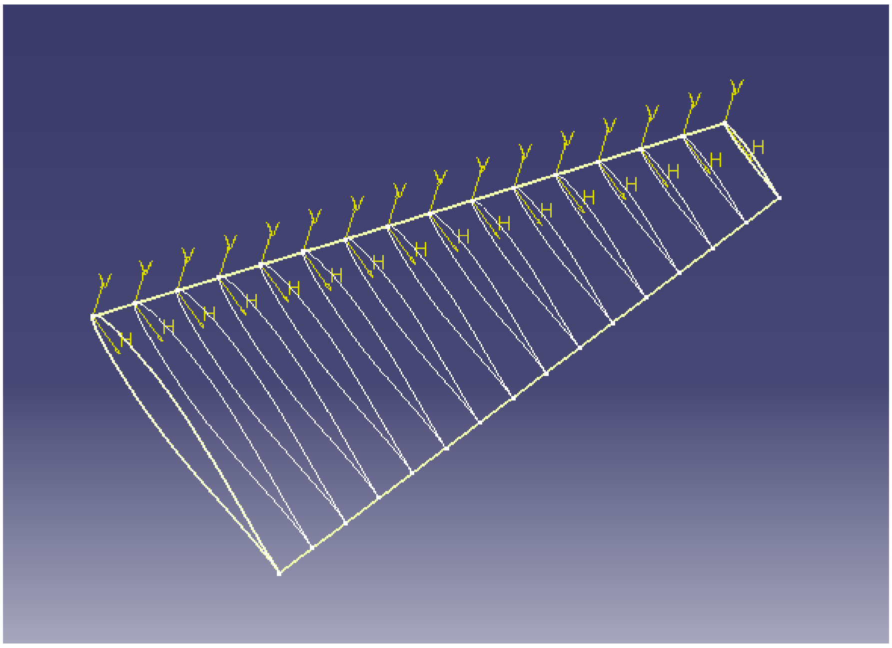

Figure 3, Sketch of the wing structure

The above figure 3 shows the span wise distribution of rib section. The rib sections are placed with the interval of 300 mm. About 750 mm of the wing structure is into the fuselage section which is completely rigid and have no degrees of freedom.

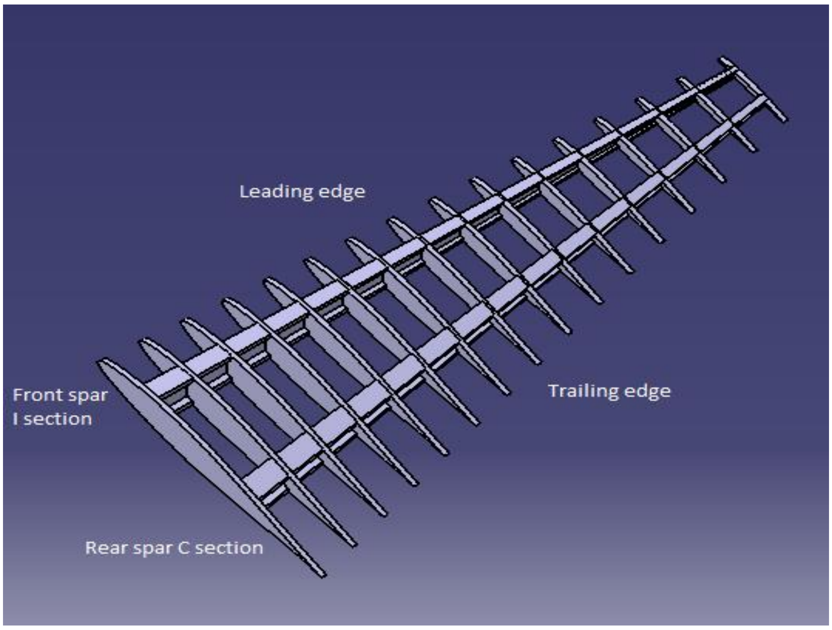

Figure 4. Isometric view of the wing structure

Figure 4, shows the isometric view of the wing assembly. Each ribs and spars are modelled separately and assembled in CATIA V5 R20.

4. LOADS ACTING OVER THE WING STRUCTURE

Lift load is considered as important criteria while designing an aircraft. Fuselage and wings are the two main regions where lift load acting in an aircraft. Here 80% of the lift load is acted on the wings (i.e., maximum lift load is acted on the wings) and remaining 20% in acted on the fuselage. Therefore in wings the maximum load is acted nearer to the wing roots. Load calculation for the wing Structure

From the basic aerodynamics,

$$L = n * W \qquad (1)$$

Where,

L = Lift produced by the entire aircraft

N = Load Factor

W = Weight of the aircraft

As we are interested to calculate the structural parameters during take-off and climbing phase, lift must be greater than weight of an aircraft.

Weight of the aircraft: 14000N

Design load factor: 6 “g”

Factor of safety: 1.5

Therefore, Total design load on the aircraft will be 126000 N.

As we mentioned earlier, total lift load on the aircraft is distributed as 80% and 20% on wing and fuselage respectively, Hence

Total load acting on the wing = 100800 N

Total load acting on the each wing = 50400 N

This force is converted into the pressure load, which is in the form of uniformly distributed load by dividing this force by the semi wing area of 8.525 m^2.Therefore, the total pressure load applied from the bottom of the surface is 5912 Pa. But we know the resultant load is acting at the distance 2138 mm (45% of from the wing root).

If one of the axes is an axis of symmetry, the bending can be determined by,

$$\sigma_b = \frac{M_Z}{I_{zz}} y + \frac{M_y}{I_{yy}} Z \qquad (2)$$

M_z - Bending moment about Z-axis. (Nmm)

σ_b - direct stress due to bending (N/mm^2)

M_y - Bending moment about Y-axis. (Nmm)

z, y- the centroid distances (mm)

Theoretically First bending frequency can be found out by using following formula

$$\text{Frequency} = \frac{1.875^2}{2X\pi X l^2}\sqrt{\frac{EI}{m}} \qquad (3)$$

Where,

l = span of the wing

E = Young's modulus

I = Moment of Inertia of the wing

m = mass/unit length

5. FINITE ELEMENT ANALYSIS

In this project MSC PATRAN software is used as the pre-processor and postprocessor. The pre-processing task includes building the geometric model by importing it from CATIA solid model of wing structure and extracting geometry, building the finite element model, giving these elements the correct material properties, setting the boundary conditions and loading conditions and finally, assembling these elements into a connected structure for analysis. Analysis is done in MSC NASTRAN solver phase. The analysis stage solves for the unknown degrees of freedom, as well as reactions and stresses. In the post processing stage, the results are evaluated and displayed. The accuracy of these results is postulated during this post processing task. The MSC NASTRAN PATRAN software together performs all 3 of the principle tasks of a finite element analysis. The consistency of the analysis is validated by refining the mesh size. The mesh size is reduced step by step until the values of the result approaches a limit.

6. MESHING OF WING STRUCTURE

The Geometry is imported into MSC PATRAN by extracting each points and curves for geometrical accuracy of the model. In wing structure there are 16 ribs and two spars. Each spar consist of two spars and a web. The following table shows the structure and the element meshed with.

TABLE II. STRUCTURES AND THE MESHING ELEMENT

Structures	**Element**
Spars	3D Tet
Ribs	3D Tet
Skin	2D quad

Special care has be taken for meshing the region in the web. Finite element properties are provided to two spars used here are I and C-section structures. While meshing top skin as well as bottom skin of the wing structure it is taken care that mesh seeds are provided for the all the positions for the later simplicity.

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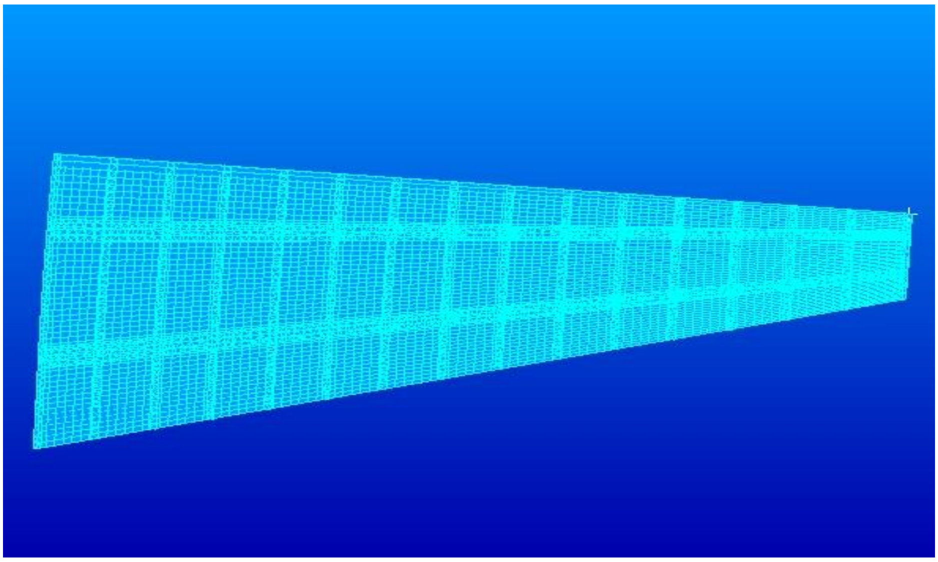

Figure 5. Meshed wing structure

7. STATIC ANALYSIS

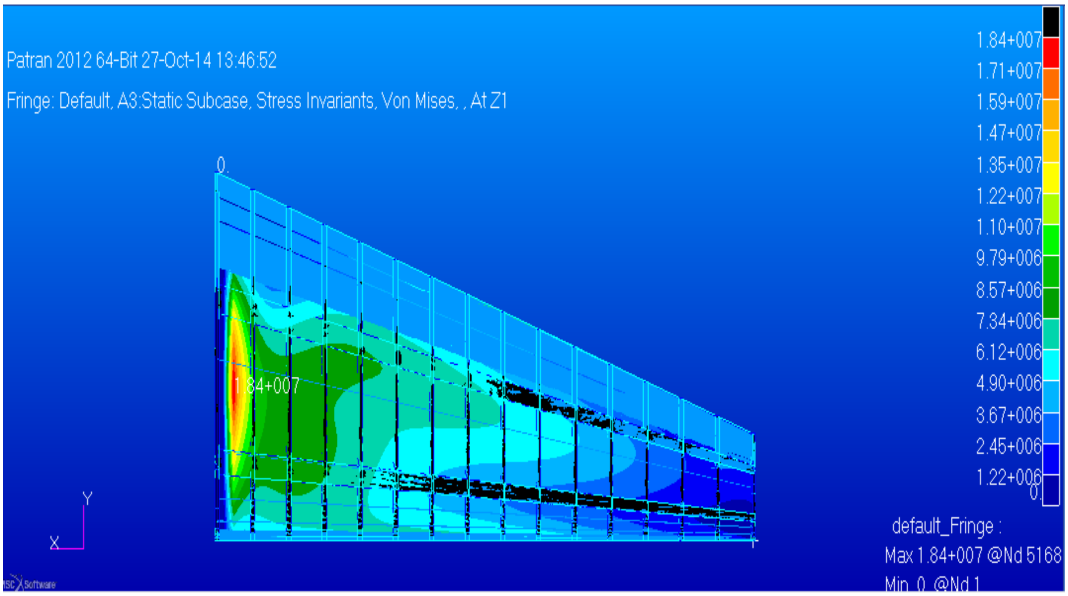

Figure 6. Equivalent Von Mises Stress distribution (Mpa)

Figure 6 shows the von mises stress (or) Equivalent tensile stress. It states that the material starts to yield when the von mises stress reaches a critical value, yield strength. The stress distribution for the given loads has been observed and that reveals the stress is distributed uniformly but maximum stresses are developed nearer to root of wing section which is shown in figure 6. In this case, the Von mises stress observed in the wing analysis is 184 Mpa which is lower than the yield strength of the AA 2024-T351. The Yield strength of the aluminium alloy AA 2024-T3 is 324 Mpa.

The structure is safe because the stress magnitude which was obtained from the analysis is less than the yield strength of the structural material.

$$\text{Factor of safety} = \frac{\text{Yield strength}}{\text{Operating load}}$$

$$= 1.76$$

The factor of safety of the Wing is 1.76 which is higher than the design factor of wing.

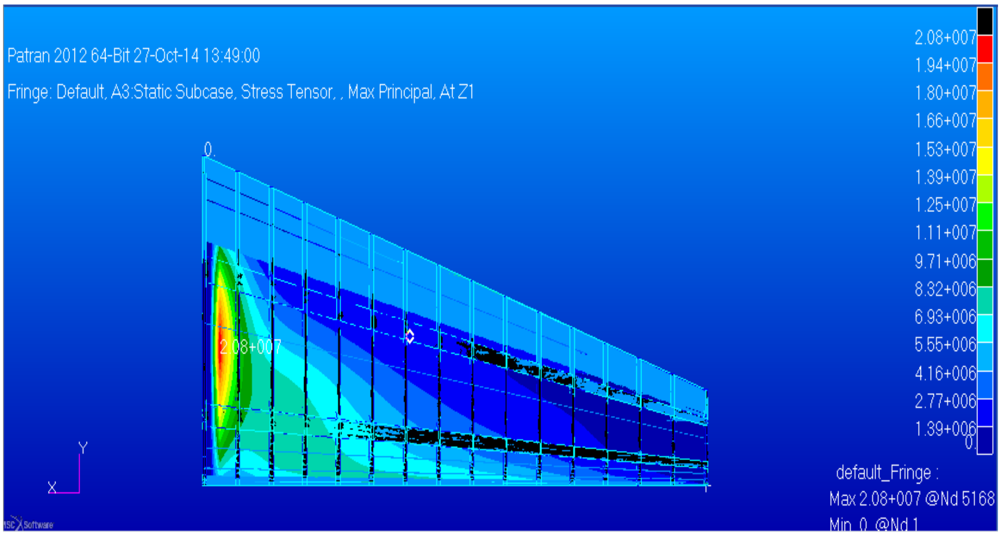

Figure 7. Maximum Principal Stress distribution (Mpa)

The above figure 7, shows the maximum principal stress distribution of the wing under the loading condition. The maximum principal stress observed in the structure is 208 Mpa which is lower than the yield strength of the material.

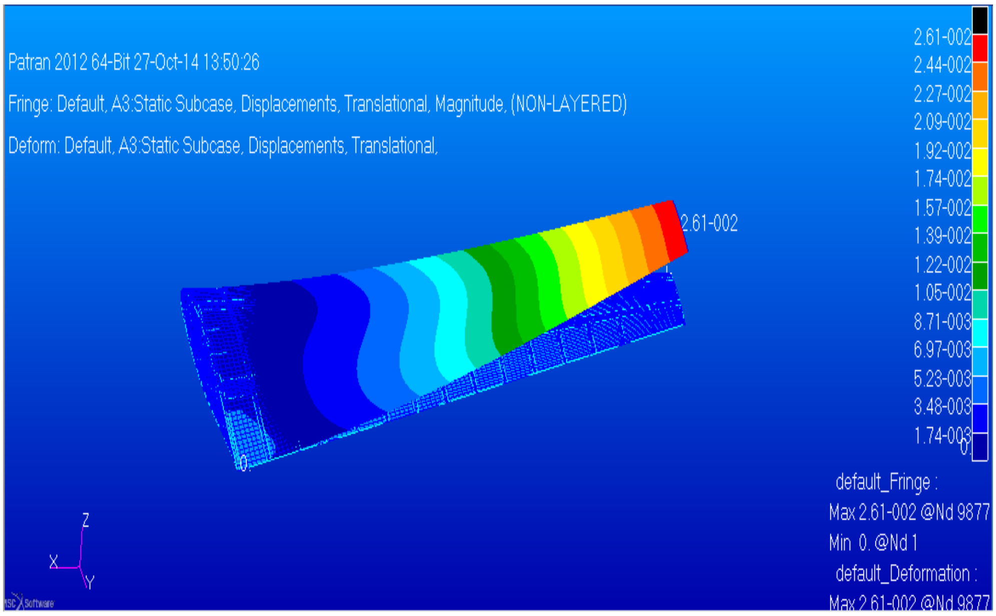

Figure 8. Deflection of the wing (m)

Figure 8 shows the deflection of the wing under the loading condition. Due to the pressure force, the wing bend upwards.

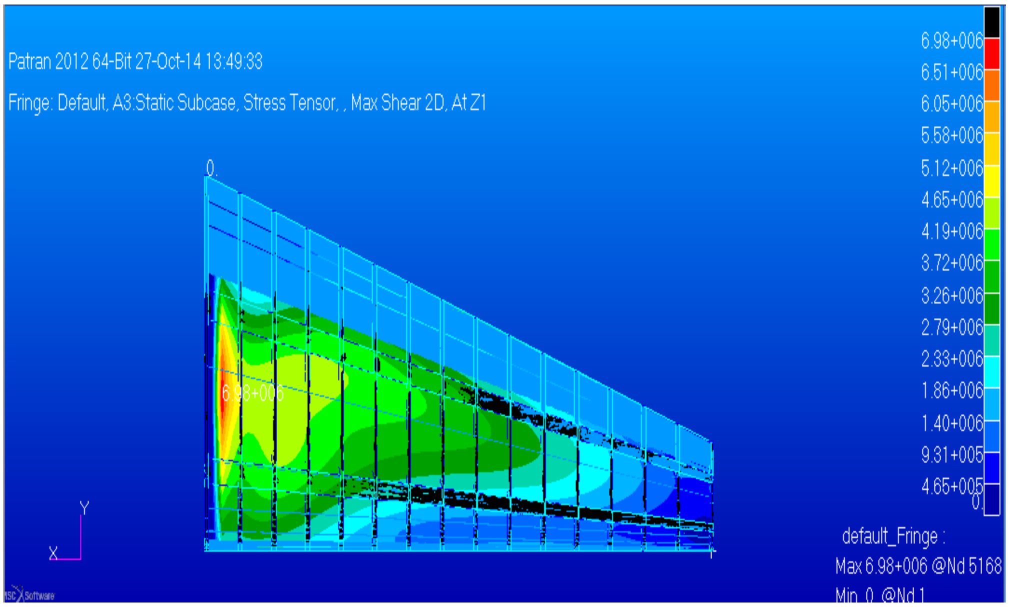

Figure 9. Maximum shear stress 2D

Figure 9 shows the distribution of shear stress in two dimension. Due to the bending of the wing, the shear stress will be more on the bottom surface of the wing. The analysis done in FEA software gives an approximate Solution. In practical there will be some variation in the result. So, correction factor is included with the result to decrease the error percentage.

Correction factor for reliability in design =0.897 [9]

TABLE III. SUMMARY OF STATIC ANALYSIS

Details	**Analysis Result**	**Corrected Result**
Equivalent Von Mises stress	184 Mpa	165 Mpa
Maximum principal stress	208 Mpa	186 Mpa
Design Factor of safety	1.76	1.57
Design load factor	6 g	6 g
Deflection	0.02 m	0.017 m

8. DYNAMIC ANALYSIS

Modal analysis is the study of the dynamic properties of structures under vibrational excitation. In aircraft due to the lift load and the load due to mounting engine on the wing leads to vibration. The modal analysis uses the overall mass and stiffness property of the structure to find the various periods at which it will naturally resonate. The types of equations which arise from modal analysis are those seen in Eigen systems. The physical interpretation of the eigenvalues and eigenvectors which come from solving the system are that they represent

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Vol.3 Issue 8, August 2015 Pgs: 1-12

the frequencies and corresponding mode shapes. The only desired modes are the lowest frequencies because they can be the most prominent modes at which the wing will vibrate, dominating all higher frequency modes.

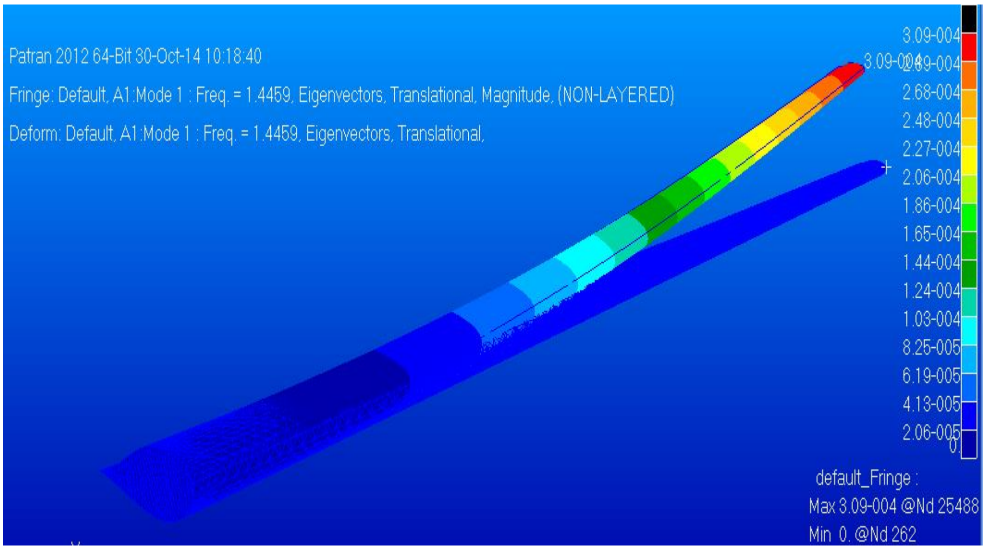

Figure 10. Mode 1

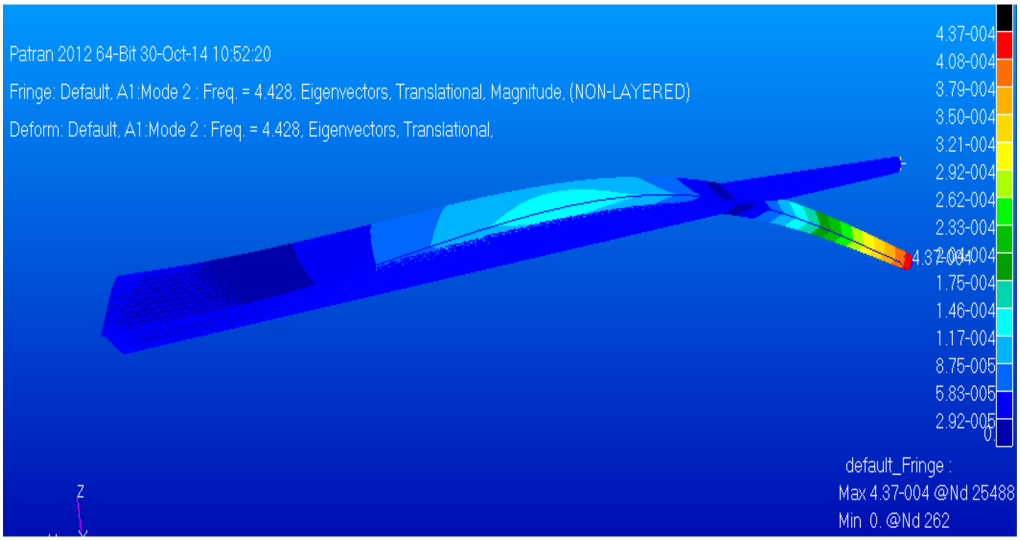

Figure 11. Mode 2

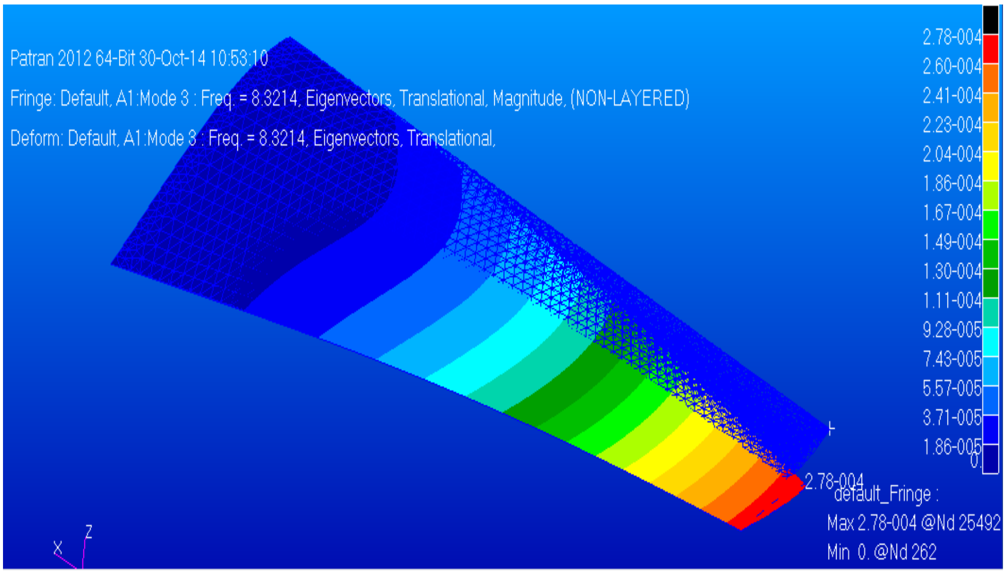

Figure 11. Mode 3

TABLE IV. SUMMARY OF DYNAMIC ANALYSIS

Mode Shape	Stress (Mpa)	Displacement (m)	Eigen Value
Mode 1	179	1	-
Mode 2	163	0.24	1.03
Mode 3	160	0.16	1.24

9 CONCLUSION

Aircraft wing model as per the plan is made in the FEA and the model is subjected to various loading. The loading given by the self-weight or due to acceleration due to gravity was discussed and the deflection over has been calculated. The wing model is severely affected by the loads on along wing direction, across wing direction, vertical direction. Moreover the combined loading is the real case. Von misses stress is calculated in order to know the maximum stress levels and minimum stress levels on the wing. Their differences are shown clearly with the contour deflections, stress levels. The deflection and stress levels are shown from minimum to maximum in the colour contours. Their values are given side by side. The wing structure has been optimally designed which satisfies the strength and stability criteria.

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