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Design and Analysis of A General Aviation Aircraft Wing Structure

Engineering

KEYWORDS : Wing Design, Aerodynamic loads, Structural Reliability.

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ABSTRACT

In the present study, design analysis is performed to accurately estimate the gross take-off weight, define the external geometry, and size the wings of a general aviation aircraft by using the performance parameters associated with a pre-defined mission profile and a set of design goals. Three-dimensional layout and projections of the design airplane are created using CATIA. In addition, finite element software package ANSYS is implemented to perform static stress analysis on the selected wing configuration when subjected to the generated aerodynamic loads to examine its structural reliability. The finite element results have shown that the selected wing configuration is a safe candidate for the present general aviation airplane implementation.

I. INTRODUCTION

The design process starts with a sketch of how the airplane is envisioned. Aircraft Wing is the complicated structure found over aircraft because of its complicated behavior towards different loads and maneuvering. The design of wings may vary according to the type of aircraft and its purpose. The aircraft wings are the primary lift producing device for an aircraft. The aircraft wings are designed aerodynamically to generate a lift force which is required in order for an aircraft to fly. Besides generating the necessary lift force, the aircraft wings are used to carry the fuel required for the mission of the aircraft, to mount engines or to carry extra fuel tanks or other armaments. The basic goal of the wing is to generate lift and minimize drag..

In modern commercial, fighter and jet aircraft, the aircraft wings are not only designed to provide the necessary lift during the different phases of flight, but also have a variety of other roles and functions. Commercial jet aircraft, the aircraft's wings are used as the primary storage system for the jet fuel required for the flight. The jet fuel is normally carried in a structure placed on the outer surface of the wing called a wing box. The fuel carried inside the wing box directly delivers fuel to the jet engines. The weight of the wing is a considerable parameter while considering the overall performance. Weight reduction of aircraft wing will increase the flight performance.

Wing structure consists of skin, ribs and spar sections. The spar carries flight loads and the weight of the wings while on the ground. Other structural and forming members such as ribs are attached to the spars, with stressed skin. 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 inspectability, and producability. Modern aircraft structures are designed using a semi-monocoque 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.

II.PROBLEM DEFINITION

In this study trainer aircraft wing structure with skin, spars and ribs is considered for the detailed analysis. The wing structure consists of 7 ribs and two spars with skin. Front spar and rear spar having I- section. Stress analysis of the whole wing section is carried out to compute the stresses at spars and ribs due to the applied pressure load.

III.OBJECTIVES OF THE PROJECT

The main objectives are: Design and structural analysis of an aircraft wing structure to compute the stresses at spars and ribs due to Pressure force over the wing section with the help of CATIA V5 and ANSYS Workbench respectively.

IV. LITERATURE REVIEW

1. A. Ramesh Kumar et al [1] Design Of An Aircraft Wing Structure For Static Analysis And Fatigue Life in his paper a wing structure of a trainer aircraft with skin, spars and ribs was considered for the detailed analysis. Stress analysis of the whole wing section is carried out to compute the stresses at spars and ribs due to the applied pressure load. The main objectives in his paper were Global and local stress analysis of an aircraft wing structure to compute the stresses at spars and ribs due to Pressure force over the wing section with the help of ANSYS Mechanical-APDL. AA 2024-T351 was used in current wing structure due to high strength and fatigue resistance properties. The ultimate tensile strength of that material was about 427 Mpa and yield strength was 324 Mpa.
2. Stress analysis of the wing structure was carried out and maximum stress was identified at wing root which was found out to be lower than yield strength of the material. Normally the fatigue crack initiates in a structure where there was maximum tensile stress was located. The fatigue calculation was carried out for the prediction of the structural life of wing structure Life of the particular region in wing structure was predicted to become critical and found out to be 3058 flying hours or 3.058 blocks, hence from the study it has advised to conduct the maintenance without fail during this period. Finally he concluded by stating that-'Fatigue crack growth analysis can be carried out in the other parts of the wing structure'.
3. Kiran Shahapurkar et al [2] Stress Analysis of an Aircraft Wing with a Landing Gear Opening Cutout of the Bottom Skin in his paper discussed about stiffened panel of a landing gear opening cutout of a typical transport aircraft wing.

FEM approach was followed for the stress analysis of the landing gear cutout of wing bottom skin .A validation for FEM approach was carried out by considering a plate with a circular hole. Maximum tensile stress of 42.99N/mm2 and maximum displacement of 2.5mm was observed in the landing gear cutout of wing bottom skin. The maximum tensile stress was acting near the rivet holes, the rivet holes were the stress raisers. A fatigue crack normally initiates from the location of maximum tensile stress in the structure, if these stresses are undetected then they may lead to a sudden catastrophic failure and result in loss of life.

4. N. S. Nirmal Raj et al [3] Static Stress Analysis For Aircraft Wing and Its Weight Reduction using Composite Material in his paper he focussed on a typical wing for business jet aircraft.

Using CATIA V5, the wing model was generated. All the dimensions are obtained by conventional methods. Loads are generated by conventional approaches. FEM was generated using MSC Nastran and Patran 2010. SOL 103 free-free run was carried out to confirm the model. SOL 101 analysis has been carried out by considering a model as metallic under given load. For modification, composite was implemented in the model aiming to weight reduction and more load bearing capability.

5. Ghassan M. Atmeh et al [4] Design and Stress Analysis of a General Aviation Aircraft Wing in his paper a general aviation airplane was designed and analyzed. A three-dimensional layout of the airplane is created using RDS software based on conic lofting, then placed in a simulation environment in Matlab which proved the designs adherence to the design goals. In addition, static stress analysis was also performed for wing design purposes.
6. Shabeer KP et al [5] OPTIMIZATION OF AIRCRAFT WING WITH COMPOSITE MATERIAL in his paper he focused to develop an accurate model for optimal design through design the structure of wing that combine the composite (Skins) and isotropic materials (all other structures) and compared this with the same wing made by changing the orientation of composite ply orientation in skin. Structural modelling was completed with the help of CATIA V5, each components moddeled separately and assembled using Assembly workbench of CATIAV5. The Von – Mises stress distribution in the case of wing is less towards the wings leading and trailing edges and decreases towards the wing tip. The variation in fiber orientation at the same skin thickness will produce the variation in the Von Mises stresses, (increase or decrease). Maximum values of Von-Mises stress was observed at the support position of the combined wing. The largest magnitude of displacement was obtained at the free end of the combined wing. The replacement of Aluminium alloy by Gr/Epoxy reduces the total weight of the aircraft wing by 23.7%. The displacement corresponding to the ply sequence [0/90/+45/-45/90/0] is of value 4.63mm and the Von Mises Stress corresponding to this sequence is 49.8N/mm2. By comparing the stress and displacement, it is concluded that the ply sequence [0/90/+45/-45/90/0] is seen to have better performance. Thus it was desirable to adopt the ply sequence [0/90/+45/-45/90/0] for composite aircraft wings in comparison with the other ply sequences considered in this study.

V. DESIGN CALCULATIONS

In this section, the practical steps for wing airfoil section selection will be presented. It is assumed that an airfoil section data base (such as NACA or Eppler) is available and the wing designer is planning to select the best airfoil from the list. The steps are as follows:

1. Determine the average aircraft weight (Wavg) in cruising flight

Wavg= (Wf++ Wi)/2 where Wi is the initial aircraft weight at the beginning of cruise and Wf is the final aircraft weight at the end of cruise.

2. Calculate the aircraft ideal cruise lift coefficient (CLc). In a cruising flight, the aircraft weight is equal to the lift force

CLc=2Wavg/ρv^2 s Where, ρ is the air density at cruising altitude S is the wing planform area

Vc is the aircraft cruise speed.

3. Calculate the wing cruise lift coefficient (CLcw).

CLcw=CLc/0.95

4. Calculate the wing airfoil ideal lift coefficient (Cli).

Cli=CLcw/0.9

5. Calculate the aircraft maximum lift coefficient (CLmax)

CLmax=2WTO/ρv2s

6. Calculate the wing maximum lift coefficient (CLmax). With the same logic that was described in step 3, the following relationship is recommended.

CLmax,w= CLmax/0.95

7. Calculate the wing airfoil gross maximum lift coefficient (Clmaxgross).

$Clmax_{gross}$= CLmax)w /0.90.

8. Select/Design the high lift device (type, geometry, and maximum deflection).
9. Determine the high lift device (HLD) contribution to the wing maximum lift coefficient
10. Calculate the wing airfoil "net" maximum lift coefficient (max Cl) Clmax=Clmaxgross-ΔClhld

VI. DESIGN APPROACH

Basing on the formulation and the type of wing to be designed. I have taken Pilatus PC7 as the reference aircraft structure to model the wing for the desired dimensions.

Pilatus pc 7	
Crew	3
capacity Passengers	0
Weight TO	26487
Area(S)	16.6
Cruise Velocity	114.4444
Stall Velocity	33.05556
Density(Cruising Altitude	0.6601
Range	2630000
Aspect Ratio	5.761952
Length	9.78
taper ratio	0.3

Based on the numerical approach defined

Parameters	Values
Average Weight	26487
Aircraft Ideal Cruise lift coefficient(CLc)	0.369109749
Wing Cruise lift coefficient(CLcw)	0.388536578
Wing Airfoil ideal lift coefficient(Cli)	0.431707309
Aircraft max lift coefficient(Clmax)	2.384128554
Wing max lift coefficient(Clmax)	2.509609004
Wing airfoil gross max lift coefficient (C_{lmax} gross)	2.788454449
High lift device	Plain flap
Delta Clhld	0.9
Airfoil net max lift coefficient (Clmax)	1.888454449

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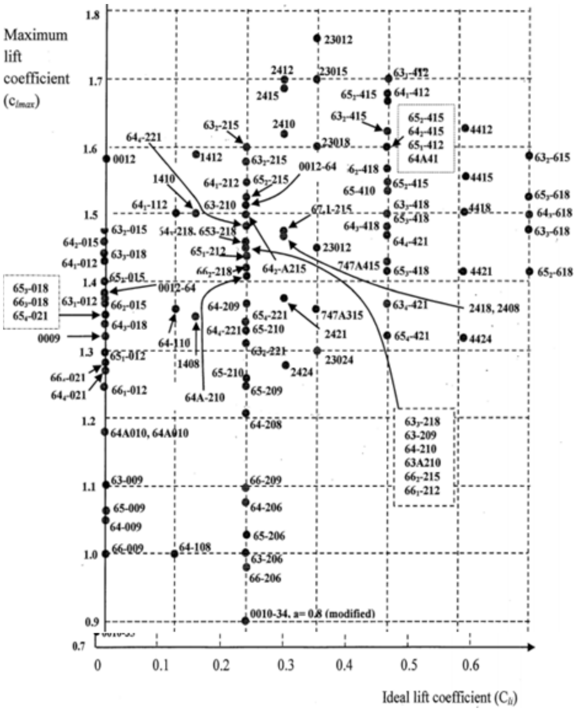

Based upon the values of Maximum lift coefficient and Ideal lift coefficient,Select the airfoil which suits for the design.**NACA 23012** is selected based upon values from the graph. The other important parameters like wing incidence angle, Mean Chord, Root Chord, Tip Chord, and Sweep Angle are calculated and are interpreted in the table below.

Parameter	Formula	Notation	Values
Wing incidence Angle		i	2
Mean Chord©	AR=b/C	C	1.69734
Root Chord	C=2/3*Cr(1+lam+lam^2/(1+lam))	Cr	2.38116
Tip Chord	lam=Ct/Cr	Ct	0.71434
Sweep Angle	Tan(^le)=((Cr-Ct)/2)/(b/2))		9.63898

1. SPAR LOCATION FROM LEADING EDGE

Table: Spar Location from Leading Edge

	Chord	Front Spar	Rear Spar	Areas
1	2380	476	1547	1812326.4
2	2094.88	418.976	1361.672	1581379.2
3	1809.76	361.952	1230.6368	1350432
4	1524.64	304.928	991.016	1119565.8
5	1239.72	247.944	805.818	888780.6
6	954.8	190.96	620.62	675864
7	714	142.8	464.1	

2. SHEAR FORCE CALCULATIONS AT SECTIONS

Table : Shear Force Calculations at Sections

	Pressure	Areas	Force	Cumulative Force
1	0.01	1812326.4	18123.264	18123.264
2	0.01	1581379.2	15813.792	33937.056
3	0.01	1350432	13504.32	47441.376
4	0.01	1119565.8	11195.658	58637.034
5	0.01	888780.6	8887.806	67524.84
6	0.01	675864	6758.64	74283.48

3. CENTROID CALCULATIONS AT SECTIONS

Table: Centroid Calculations at Sections

	A	b	H	Centroid
1	2380	2094.88	810	413.601616
2	2094.88	1809.76	810	414.857810
3	1809.76	1524.64	810	416.543666
4	1524.64	1239.72	810	418.914323
5	1239.72	954.8	810	422.527386
6	954.8	714	810	424.479865

4. LOAD DISTRIBUTIONS ON SPARS

Table: Bending Moments at Sections

	BENDING MOMENT	BENDING MOMENT(Nm)	B.M on front spar	B.M on rear spar
1	7495811.28	7495.81128	3440577.3	4055233.90
2	6560475.12	6560.47512	3011258.0	3549217.04
3	5625138.96	5625.13896	2581938.7	3043200.17
4	4690021.5	4690.0215	2152719.8	2537301.63
5	3755341.44	3755.34144	1723701.7	2031639.71
6	2868906.6	2868.9066	1316828.1	1552078.47

5. MOMENT OF INERTIA OF SPARS

Table: Moment of Inertia of Spars

	FRONT SPAR			
	Chord	height(a)	Y	M.O.I
1	2380	274.9	137.45	1107511.3
2	2094.88	247	123.5	870937.64
3	1809.76	212	106	640949.67
4	1524.64	179.5	89.75	452474.49
5	1239.72	146.28	73.14	295249.51
6	954.8	112.62	56.31	173654.78
	REAR SPAR			
	Chord	height(b)	Y	M.O.I
1	2380	194.39	97.195	923064.30
2	2094.88	171.09	85.545	711048.64
3	1809.76	147.88	73.94	526965.38
4	1524.64	124.52	62.26	369958.78
5	1239.72	101.23	50.615	240823.05
6	954.8	77.94	38.97	141649.87

	Cp	h (dist b/w fs & rs)
1	1071	1071
2	942.696	942.696
3	814.392	814.392
4	686.088	686.088
5	557.874	557.874
6	429.66	429.66

7. TORQUE ACTING DUE TO LOAD ON SECTIONS

Table: Torque due to Load on Section

	a on RS	b on FS	H	A	CG of Trape-zium	FS	CG from LE	CP
1	194.39	274.9	87.47	20525.68	0.636	476	476.63	1071
2	171.09	247	76.9905	16094.47	0.634	418.976	419.61	942.696
3	147.88	212	66.546	11974.28	0.634	361.952	362.58	814.392
4	124.52	179.5	56.034	8517.728	0.634	304.928	305.56	686.088
5	101.23	146.28	45.5535	5637.473	0.634	247.9	248.57	557.874
6	77.94	112.62	35.073	3341.755	0.6340	190.96	191.5940523	429.66

	D	Load on Section(N)	Torque(Nmm)	Cumula-tive T	Shear Flow q	Cumula-tive q
1	594.3636	18123.26	10771808	10771808.4	262.3982	262.3982
2	523.0859	15813.79	8271970.8	19043779.3	256.9816	591.6246
3	451.8051	13504.32	6101320.4	25145099.79	254.7676	1049.962
4	380.5257	11195.66	4260235.4	29405335.27	250.0805	1726.125
5	309.296	8887.806	2748962.4	32154297.7	243.8116	2851.836
6	238.0659	6758.64	1609002.0	33763299.74	240.742	5051.731

8. SHEAR FLOW DUE TO TORSION

- SHEAR FORCE due to q=a*cumulative q
- SHEAR FORCE due to q=45.9% of SF due BM
- Total Shear Force = shear force due to q+ shear force due to B.M

9. WEB THICKNESS FOR SHEAR STRENGTH OF 283 N/mm2

Using below formulae, calculate web thickness.

- WEB THICKNESS, t = total shear force/(b*283)
- A= b*t
- MOI of web = (tb^3)/12
- MOI of front spar= moment of inertia=(bending moment * Y)/427
- MOI of flange =MOI of F.S – MOI of Web
- A flange =√((MOI of flange)/(Y¯ of FS))
- A of FS= AFLANGE+AWEB
- VOLUME of FS= A of FS*810

10. BUCKLING ITERATIONS

- F induced =q/t web
- Front spar -- t =b*SQRT(Finduced /(5*72400))
- Rear spar --t =a*SQRT(F induced /(5*72400))

VII. MODELING PROCEDURE

1. DESIGN SPECIFICATIONS

Table: Design Parameters

Thickness of web		Height of the web		Thickness of flange		Height of the top flange	
FS	RS	FS	RS	FS	RS	FS	RS
3.57	2.84	274.90	194.39	5.41	4.30	77.26	87.69
4.37	3.42	247.00	171.09	6.63	5.17	57.29	64.71
4.77	3.75	212.00	147.88	7.23	5.69	46.41	51.56
5.04	3.95	179.50	124.52	7.64	5.98	38.53	42.08
5.20	4.07	146.28	101.23	7.87	6.16	32.24	34.34
5.28	4.13	112.62	77.94	8.00	6.26	27.09	27.85

A flange	A flange	A Top flange	A Top flange	Location of Spar	
552.24	497.50	276.12	248.75	476.00	1547.00
501.18	441.99	250.59	221.00	556.51	1499.21
442.88	387.01	221.44	193.51	637.03	1505.71
388.37	332.39	194.18	166.19	717.54	1403.63
335.02	279.25	167.51	139.62	798.15	1355.97
286.18	230.17	143.09	115.09	878.86	1308.31
				973.122	1294.422

2. MODELING OF WING

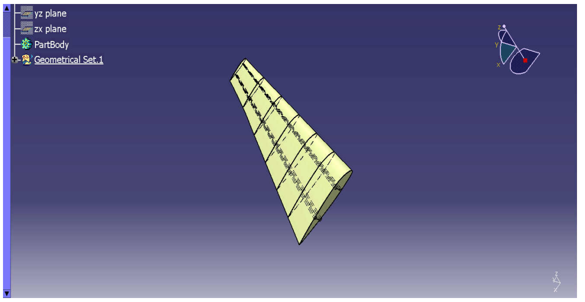

Fig: Complete wing with a skin over

VII.COMPUTATIONAL ANALYSIS

1. ANALYSIS PROCEDURE

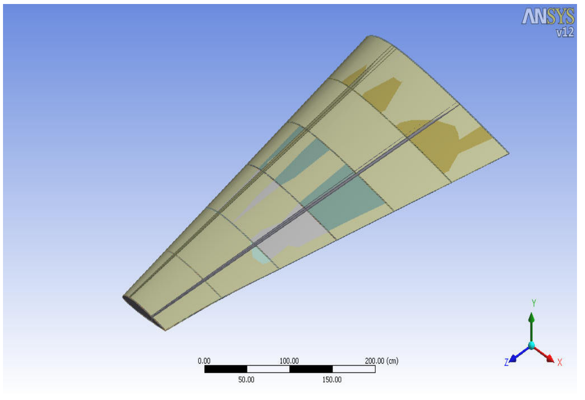

Fig: Imported wing structure with skin

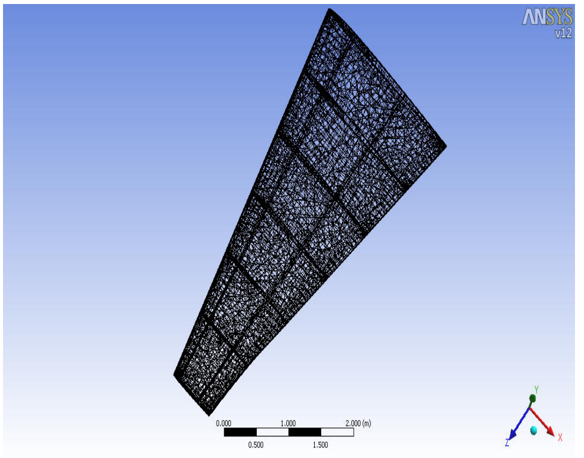

Fig: Meshed model

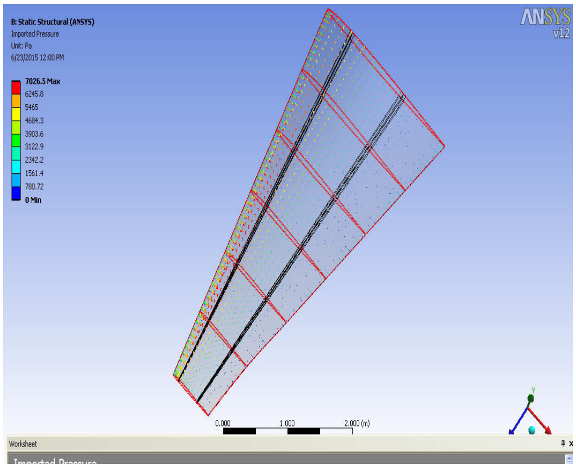

Fig:Imported Pressure on the wing structure

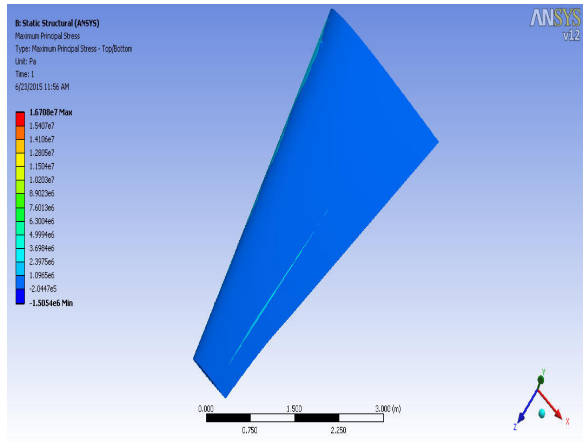

Fig: Maximum principal stress on the wing

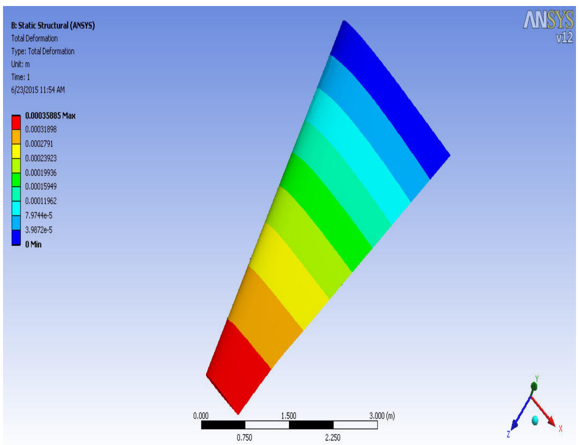

Fig:Total deformation of the wing.

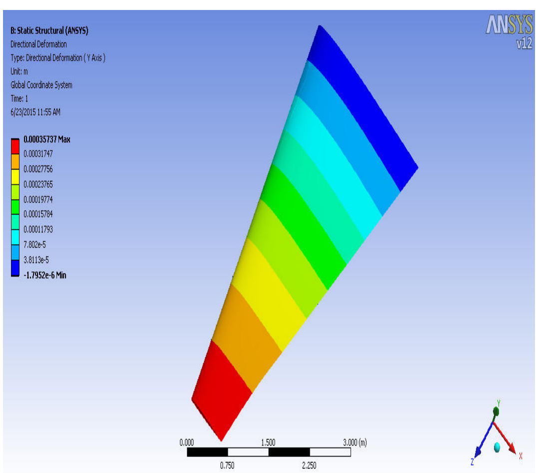

Fig:Directional deformation of the wing

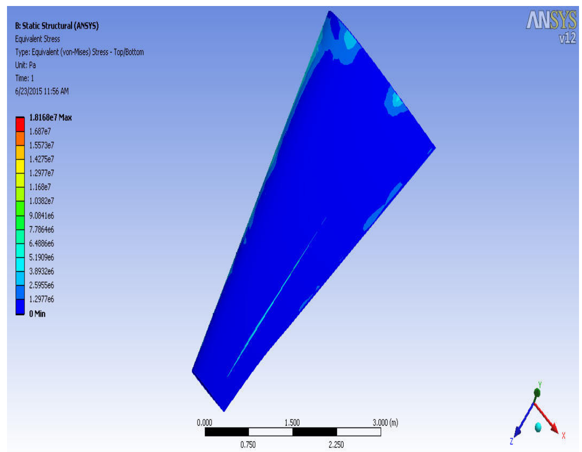

Fig: Von-mises stress on the wing structure

VIII.RESULTS AND DISCUSSIONS

S.NO	RESULTS	VALUES
1	Total Deformation	0.3585 mm
2	Von-Mises Stress	181.66 MPa
3	Max.Principal Stress	167.08 MPa

Factor of safety: It is defined as the ratio of ultimate strength (yield strength) to working strength.

- The ultimate tensile strength of Aluminium alloy(AA2024 T6) is 427 MPa.
- The working strength of the material is found to be 181.66MPa
- Hence, F.O.S (factor of safety) of the considered material for the wing is calculated and found to be 2.35 which is greater than 1.5.

IX.CONCLUSION

- The total deformation of the wing due to applied loads is 0.3585mm.
- The maximum principal stress of the wing due to applied loads is 167.08MPa.
- The equivalent or Von-Mises stress due to applied loads is 181.66Mpa.
- F.O.S (factor of safety) of the Aluminium alloy(AA2024 T6) is 2.35.
- From the above obtained results, we can say that the material which we have taken is capable of bearing high stresses as it is evident from the F.O.S and it can be used for General Aviation Aircrafts in the future generations.

- This project helped in understanding various software's and analyzing it under various conditions and different stresses.
- The meshing of a complex part is made simple by keenly observing and applying the known dimensions, which is a tricky job.
- Through the project the applications of the tools present in the software has become easy and all the complex parts are being made appropriately with ease.

FUTURE SCOPE

- There is always an improving technology arising in the field of aeronautical. The wing concept is also one such idea where the design gets changing frequently to improve the performance of the aircraft.
- Many technologies have been developed over the years to meet the challenges of wing design and development. These technologies have matured over the years and widely used in the current wing structure and new technologies will continue to evolve in future.
- The additional study that can be done apart from the work submitted is the flow analysis on the main wing through Computational Fluid Dynamics methods.
- The extension of the project may be done by applying or using various materials for the aircrafts wing. By introducing various materials and performing various tests the strength of the each material or composite is known and is applied to the main wing. The point of interest is that the fighter aircraft should be stronger to resist any kind of load at any moment as they may face different and difficult task depending on the situation.
- The future wing for design aircraft poses many new challenges in configuration design, use of materials, design and analysis methods. These challenges can be met, while adhering to all regulatory requirements of safety, by employing advanced technologies, materials, analysis methods, processes and production methods. By applying functional simulation and developing design tools, the development time and cost reduced considerably.

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