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Performance Evaluation of Aircraft Wing for Different Lamina Orientation of Laminated Composite

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ABSTRACT

An aircraft is a complex structure, but very efficient man-made flying machine. Aircraft are generally built up from the basic components of wings, fuselage, tail units and control surfaces. Each component has one or more specific functions and must be designed to ensure that it can carry out these functions safely. Any small failure of any these components may lead to catastrophic disaster leading to huge destruction of lives and property. If the structural integrity of these components is achieved as designed, it means that the structure has the capacity to retain its desired aerodynamic shape under flight conditions.

The optimum structural design of an Air craft 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 maneuvering situations.

The critical element of aircraft is the design of the wings. Several factors influence the selection of material of which strength allied to lightness is the most important. Composite materials are well known for their excellent combination of high structural stiffness and low weight. Because of higher stiffness-to-weight or strength-to-weight ratios compared to isotropic materials, composite laminates are becoming more popular.

The objective of this project is to perform static structural analysis to evaluate stresses and displacement induced in air craft wing for different lamina orientation. Also to study the effect of stacking sequence and lamina thickness on performance of wing structure.

Keywords - *Composite Wing Structure, Finite Element Analysis in Ansys, Optimum Ply Orientation, Static Structural Analysis.*

1. INTRODUCTION

The original of traditional controlled air ship was built of wood and canvas. At that point aluminum and composite materials were utilized as a part of flying machine development. The expanded wing loadings and complex basic types of present day flying machines cause high stretch fixations for which the ordinary material is not all around adjusted. Presently a day, the carbon fiber composites have supplanted the customary metals. FRCs are regularly utilized as a part of the aviation, car, marine, and development commercial enterprises. Carbon fiber strengthened polymer or carbon fiber fortified plastic or carbon fiber strengthened thermoplastic (CRP, CFRP or regularly essentially carbon fiber), is a to a great degree solid and light fiber strengthened polymer which contains carbon strands. Utilization of CFRP makes the air ship lighter with included advantage of less upkeep, super weariness resistance and high fuel productivity. These composite materials can give a much higher quality to weight proportion and firmness to-weight proportion than metals, now and then as much as 34% to 40% better [1]. These CFRP composites are comprised of fiber layers or overlays of various utilize introduction. The goal is to lead parametric study on composite flying machine wing by changing the employ introduction to locate the comparing variety in anxiety and relocation. This study did a preparatory auxiliary outline and examination on fundamental wing, control surface and joint parts.

Composite structure is surely understood meant for excellent blend of high basic firmness and less in weight. In light of higher solidness to-weight or quality to-weight proportions contrasted with isotropic materials, so composite covers are turning out to be more famous. Composite structures normally comprise of overlays stacked with layers of various fiber introduction points. The layer thickness is typically settled, and fiber introduction edges are frequently constrained to a discrete set, for example, 0°, ±30°, ±45°, ±75°, and 90°. This prompts distinctive blends of handle introduction moreover among those one will give the healthier results, which is the improved configuration for composite structures.

In composites unidirectional overlay is a cover where all strands are arranged in one bearing, cross-handle overlay is a cover were the layers of unidirectional lamina are situated at exact points with one another and semi isotropic overlay acts comparably like isotropic material, here the versatile properties are identical in all courses. Unidirectional composite structures are adequate just to carry straightforward loads, for example, uniaxial strain or unadulterated twisting. In composites structures with complex prerequisites of stacking and solidness, structures including edge handles will be fundamental. Ever since every cover within the composite structure contain unmistakable fiber introductions which may change from the connecting covers, the ideal utilize introduction is additionally acquired as an after effect of the parametric study directed utilizing limited component bundle by differing the introduction arrangement in the composite.

The result of the auxiliary investigation demonstrated that the handle introduction significantly affect the mechanical execution of the composite overlays such that different outline destinations can be accomplished just by selecting the best possible game plan of employ introduction and thickness.

2. GEOMETRICAL CONFIGURATION OF AIRCRAFT WING

The aircraft wing design is a complex iterative procedure in addition the calculations are commonly repeated quite a lot of times. Wide variety of design tools and software like comsol were used to develop a optimized wing structure considering aerodynamic principles in addition several numerical techniques were developed in past to reduce the complexity iterations for development of desired structure[3]. After multiple iteration the outcome is a two spar construction of wing design is suitable for commercial transport air vehicle. Here in this design the spar at the leading edge of the wing is known as front spar and the spar near to end portion of the wing is called rear spar. Extreme end of the spar is mounted to fuselage is known as root of wing and other side towards the tip of wing is called free end. This type of arrangement is identical to simple cantilever beam problem in every engineering structure.

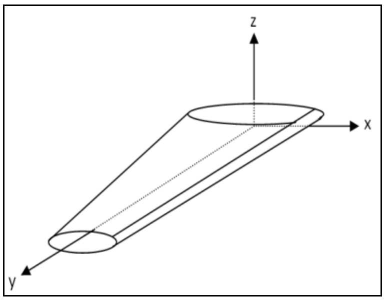

Fig 1: Outline Structure of Aircraft Wing in Local Coordinate System.

2.1 Conceptual Design of Wing Structure

Aircraft wing design is difficult and prolonged system of work which mainly divided in 3 phases.

- Conceptual design
- Preliminary design
- Detailed design

Now Conceptual design of aircraft wing is mainly considered, that's the umbrella of current research were it deals mainly the external geometry of wing structure includes primary structure, components and the material used to optimize the weight and determining the performance characteristics that the aircraft wing must have with a purpose to attain its design desires. Generally designing of wing structure is important criteria in lift load consideration were 80 % of the load is taken by the wings of aircraft and the remaining 20% is by the fuselage.

2.2. Aerodynamic Loads

Motion of complete aircraft mainly affected by the aerodynamic forces i.e. the drag force and lift forces, when it travels all the way throughout the atmosphere. Wing is the main source of this aerodynamic force and wing is that surface of plane which helps the aircraft by using the dynamic reaction at the air generating stress distribution. This stress distribution on the wing structure will going to change at every angle of point and flight situation. Therefore aircraft manufacturing industries use modern technique and come out with a idea of semi-monologue construction were the load carrying structure is armored with longerons and frames inside the structural body of the aircraft and the surface of the wing is wrapped with composite skin and main loads are taken by ribs and spars of the wing structure [4]. Solving aerodynamic loads is a very complex iteration process in redundant structures so programmed matrix method is used to solve the internal loads of the structure and now present finite element models of wing structure involves thousands of nodes and for each node there are more no of degree of freedom which will give the detail information of the part and this leads to determine the required dimensions of the structure to withstand the deflections, buckling, stress and strain levels.

3. GEOMETRY AND FE MODELING

The aircraft wing structure is analyzed for its structural behavior. The linear static structural analysis is carried out on aircraft wing to evaluate the stresses, displacement induced in the aircraft wing components for the gravity and wind pressure load cases. In this project work, analysis is carried out on wing structure to study the effect of thickness change and lamina stacking sequence on structure behavior of aircraft wing assembly.

3.1 Geometry

The aircraft wing assembly considered for the analysis is shown in fig 2.

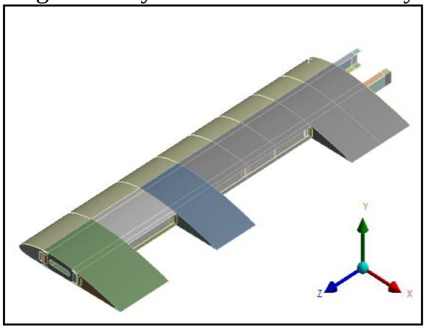

Fig 2: Aircraft wing Assembly.

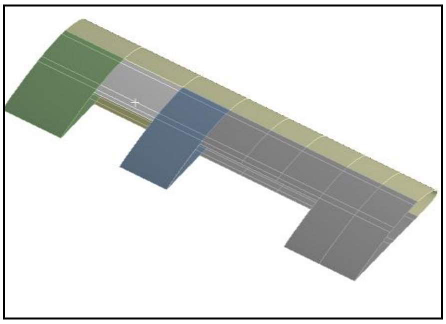

Fig 3: Wing Skin.

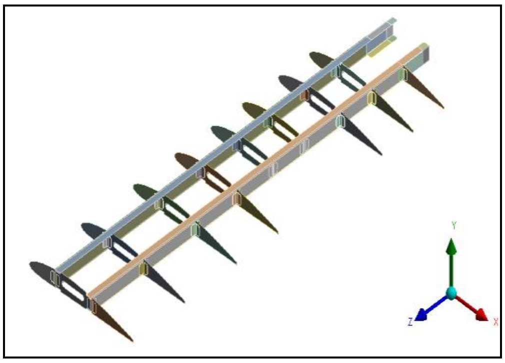

Fig 4: Ribs, Bracket & Spar Assembly.

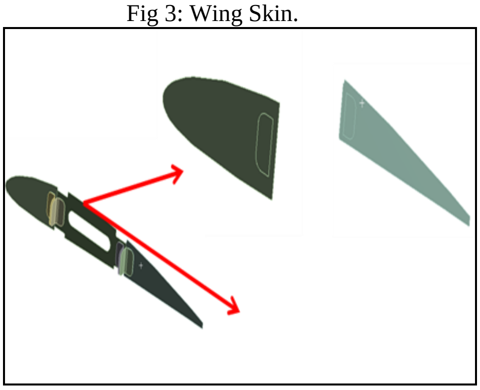

Fig 5: Leading Edge & Trailing Edge.

The wing components gauge thickness, material details are listed in the below table 1.

Component	Material	Thickness (mm)
Spars	Aluminum 7075 - T651	2.54
Leading Edge Ribs	Aluminum 7075 - T651	0.8
Trailing Edge Ribs	Aluminum 7075 - T651	0.8
Brackets	Aluminum 7075 - T651	1.65
Composite Skin	Carbon Epoxy composite	1.5

Table 1: Wing Component Material and Thickness Details.

3.2 Finite Element Modeling

The FE model is generated using meshing software ANSYS Workbench V15.0. The 2D shell mesh is used for FE modeling of all wing components. The FE model aircraft wing is modeled using SHELL 181 element.

The FE model is created with a global element size of 15mm with capturing all features. The FE model is shown in Fig 6 & 7.

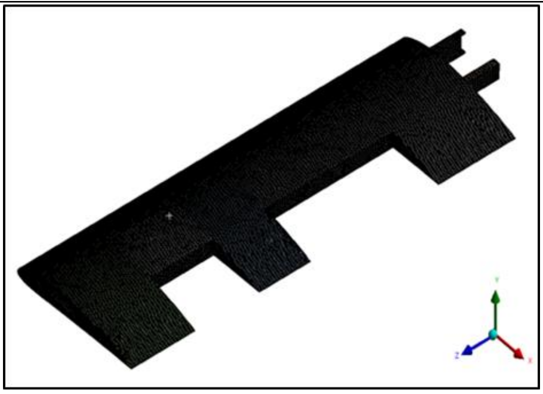

Fig 6: Wing Assembly FE model.

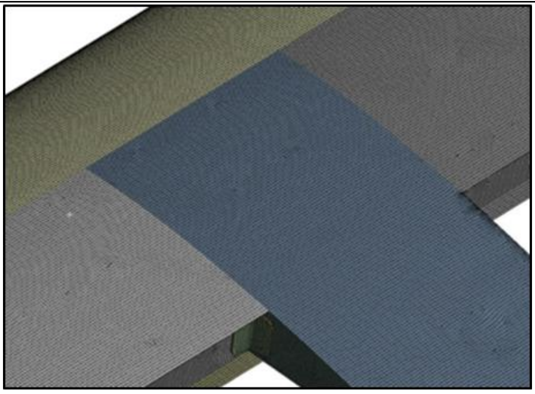

Fig 7: Wing Skin FE model.

3.3 Material Properties

The wing assembly is analyzed for 4 different materials for different stacking sequences. The material properties considered for the analysis is listed below.

CFR Epoxy composites-LY3/1

The Skin of aircraft wing is modeled using CFR – Epoxy composite material. The Epoxy resin Araldite LY 5052 is used as resin material. The UD carbon fiber Torayca T700 used.

Carbon Epoxy Composite Material		
Properties	Values	Unit
Young's Modulus along fiber direction1 (E11)	67.4	GPa
Young's Modulus along matrix direction2 (E22)	17.73	GPa
Young's Modulus along matrix direction3 (E33)	17.73	GPa
Poisson's ratio (v12)	0.47	
Poisson's ratio (v23)	0.40	
Poisson's ratio (v13)	0.47	
Shear modulus in 1-2 plane (G12)	11.34	GPa
Shear modulus in 2-3 plane (G23)	11.34	GPa
Shear modulus in 1-3 plane (G13)	11.34	GPa
Density	1490	Kg/m^3

Table 2: Material Properties for Wing Skin.

The aircraft wing skin is made of composite material and spar assembly is made of aluminum 7076 T651 grade material. The material properties used for the analysis is listed in table 2.

Aluminum 7075 T651		
Properties	Values	Unit
Young’s Modulus (E)	71.7	GPa
Shear Modulus (G)	26.9	GPa
Poisson's ratio (v)	0.33	
Ultimate Strength	572	MPa
Yield Strength	503	MPa
Shear Strength	331	MPa

Table 3: Material Properties for Spar Assembly Components.

3.4 Layer Definition

The composite laminate has 3 layers. The thickness of each layer is 0. 5mm and total thickness of laminated composites is 1.5mm. Lamina stacking sequence is [0/35/0], layer definition is shown in fig 8.

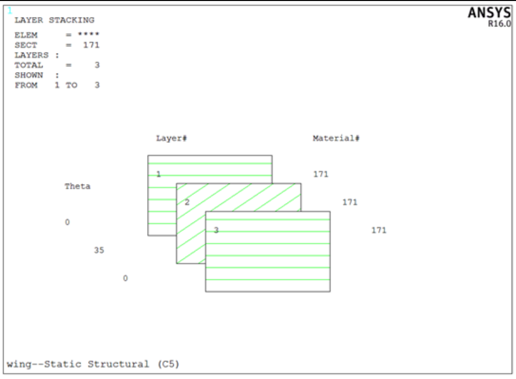

Fig 8: Layer Stacking.

4. RESULTS AND DISCUSSIONS

The aircraft wing is analyzed for gravity and wind air pressure loads under different materials by changing lamina stacking sequences. The boundary conditions and load data are listed below. The static structural analysis is carried out on wing structure to determine the displacements and stresses induced in the various parts of wing assemblies like skin, ribs, spars and brackets. The results are discussed below.

5.1 Load and Boundary Conditions

The aircraft wings are majorly exposing to gravity loads and wind air pressure loads. The aircraft wing is attached to fuselage at one of its extreme end and other end is exposed to wing air pressure. Hence one end of the wing is constrained for all DOF as shown in fig 9. The gravity load of 13906 mm/s^2 is applied on complete wing structure in vertically down ward direction. The wind pressure load of 0.00295MPa is applied on wing skin (Probable wind exposing area) as shown in fig 10.

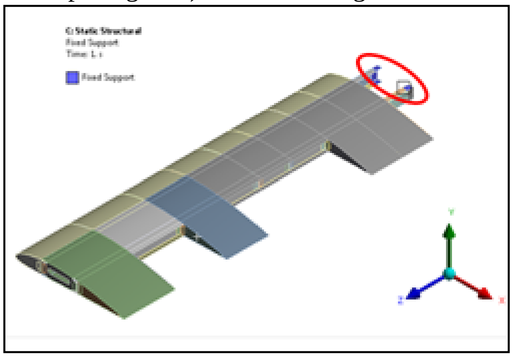

Fig 9: Boundary Conditions for Analysis.

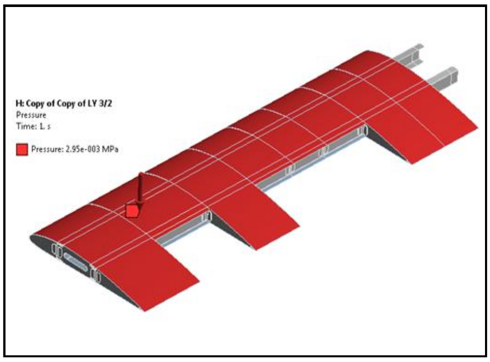

Fig 10: Wind Air pressure Load on Wing Structure.

5.2 Results

Iteration-01:- Material 1: CFR- Epoxy Composites – LY 3/1.

The static structural analysis is carried out for Material 1 i.e. Carbon Epoxy material with stacking sequence of [0/30/0] with 3 layers of lamina having thickness of 0.5mm/lamina. The result for this material is discussed below.

The displacement induced in the wing structure is shown in fig 11 to 12.

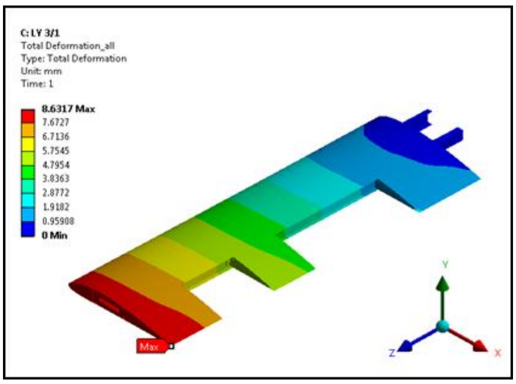

Fig 11: Displacement plot for Wing Assembly.

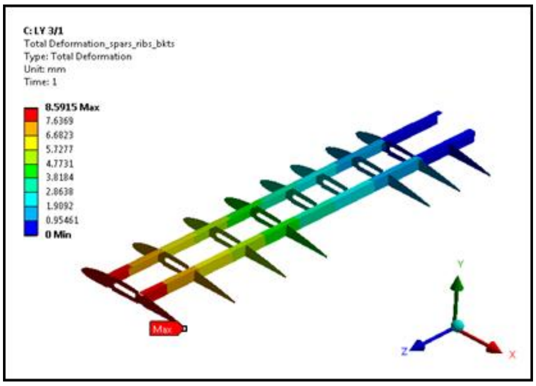

Fig 12: Displacement plot for Wing Skin.

The maximum displacement induced in the wing assembly is 8.59mm observed at extreme end of the wing assembly. The wing mounting on aero plane resembles like cantilever beam, from Strength of material it is clear that the maximum displacement for cantilever bending will be at end, hence induced maximum displacement is in line with the real scenario.

The normal stress distribution in X & Y direction in the wing skin is shown in fig 13 to 16.

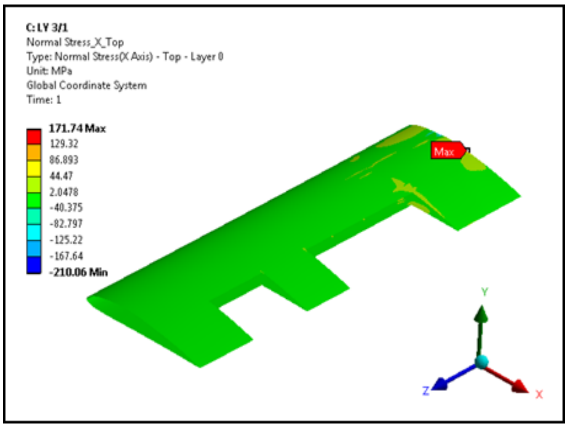

Fig 13: Normal stress X direction Top Layer

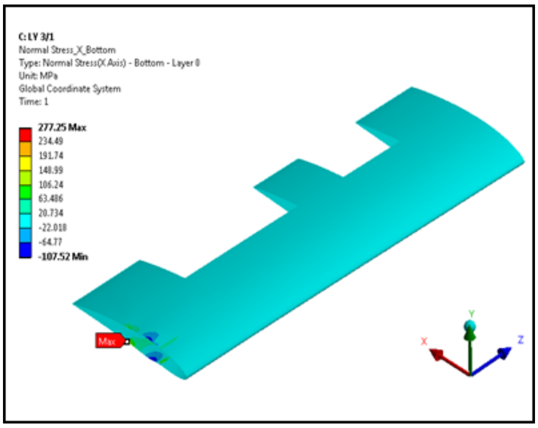

Fig 14: Normal stress X direction Bottom Layer.

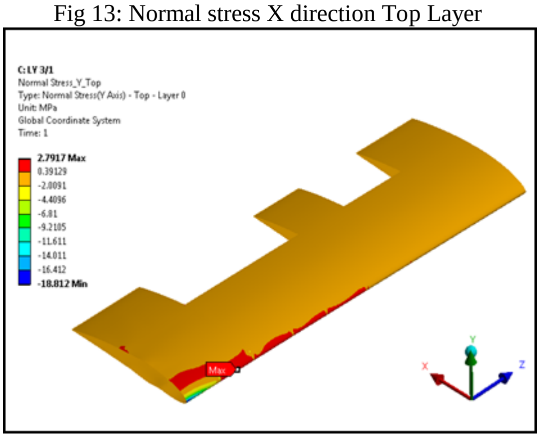

Fig 15: Normal Stress Y direction Top Layer

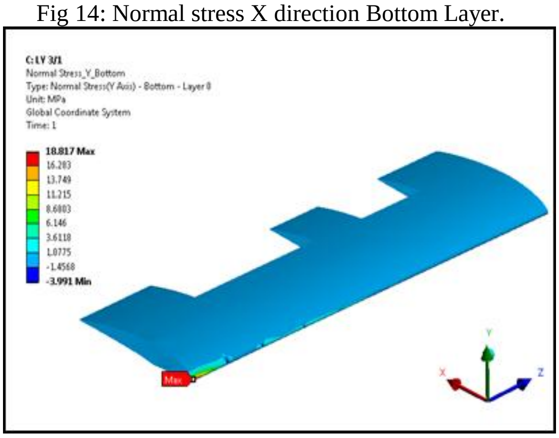

Fig 16: Normal Stress Y direction Bottom Layer.

The wing skins are made of laminated composite structure, laminated composite structures are likely to fails under normal directional stresses rather than von-Mises stresses, hence for wing structure normal stresses are listed out. The maximum normal stress in X direction is 277MPa and maximum normal stress in Y direction is 19MPa.

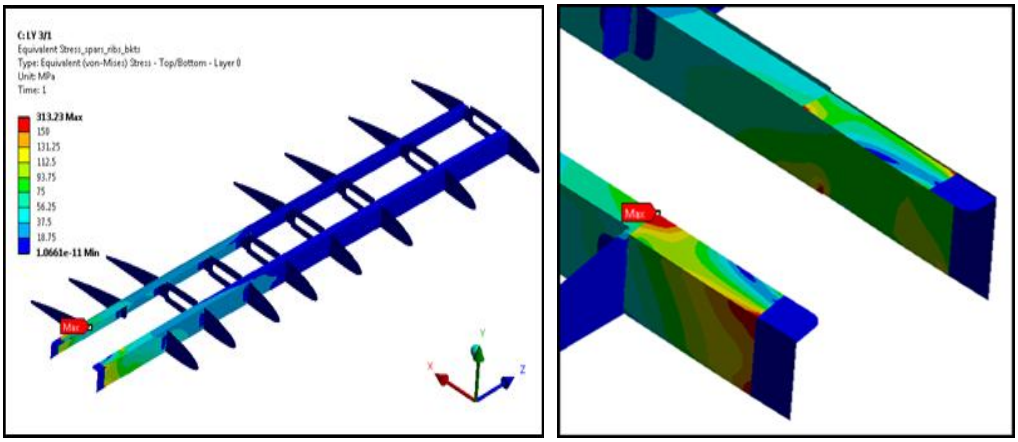

Fig 17: von-Mises Stress on Spar Assembly.

The bracket, ribs and spars assembly components are made of aluminum alloy. The induced von-Mises stress is compared with the yield limit of the material to determine whether component is safe or not. The maximum von-Mises stress induced in the Rib is 37MPa, in bracket is 41MPa and in spar assembly is 313MPa.

5.3. Discussion

The normal stress and displacement induced in the wing structure for different lamina stacking sequence is listed table 4.

Material	Lamina Stacking Sequence	Lamina Stacking Sequence	Normal Stress - Wing Skin (MPa)		von-Mises stress (MPa)			Displacement (mm)
			X Direction	Y Direction	Brackets	Ribs	Spars	
Carbon Epoxy Composite	LY 3/1	0/35/0	277.25	18.81	41.49	37.27	313.23	8.63
	LY 5/2	0/-35/0/35/0	318.79	25.42	35.00	30.00	319.74	7.16
	LY 5/3	0/90/0/90/0	256.83	6.16	26.49	19.05	302.07	7.69
	LY 11/4	0/-35/45/90/-45/0/45/90/-45/35/0	208.10	17.49	28.87	25.70	327.93	6.12

Table 4: Normal Stress and Displacement induced in the wing assembly.

The variation of normal stress in X direction w.r.t lamina stacking sequence is shown in fig 18.

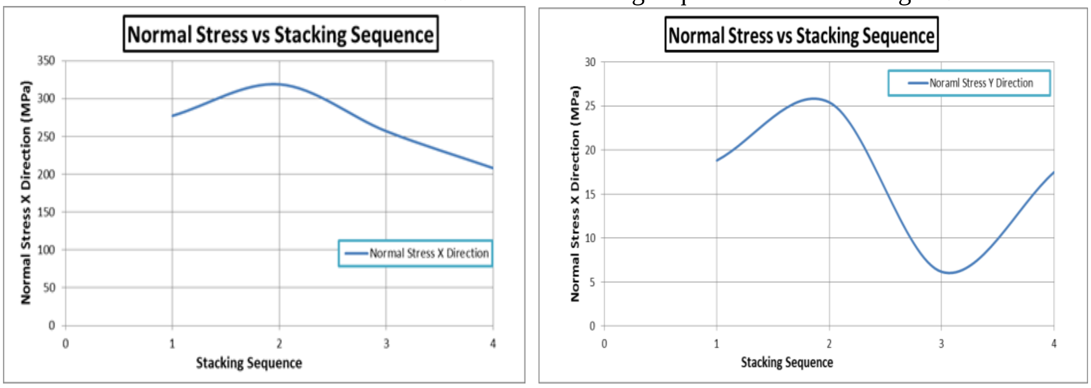

Fig 18: Normal Stress X & Y direction vs stacking sequencing for Wing Skin.

From the graph it is clear that, normal stress in X direction is decreases with increase in number of lamina. The stacking sequence LY5/2 is having more normal stress of 319 MPa and stacking sequence LY11/4 is having minimum normal stress of 208MPa. The combination of 35^0, 45^0 and 90^0 lamina gives the better performance compare to other lamina angles.

The stacking sequence LY5/2 is having more normal stress of 25 MPa and stacking sequence LY5/3 is having minimum normal stress of 6MPa. The variation of normal stress in Y direction is less among all stacking sequences.

The variation of von-Mises stresses induced in the rib and bracket w.r.t lamina stacking sequence is shown in fig 19.

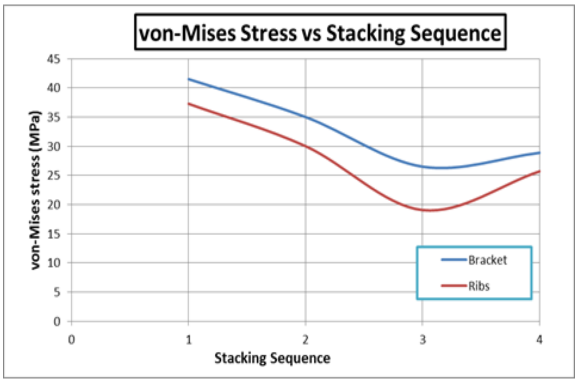

Fig 19: von-Mises Stress for Bracket and Ribs.

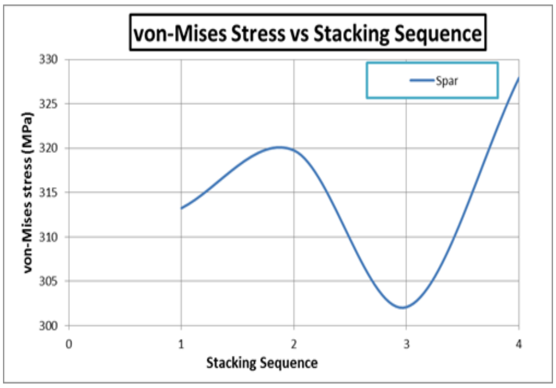

Fig 20: von-Mises Stress for Spar.

From the graph it is clear that, von-Mises stress is decreases with increase in number of lamina. The stacking sequence LY3/1 is having more von-Mises stress of 41MPa & 37MPa in bracket and ribs respectively and Stacking sequence LY11/4 is having minimum von-Mises stress of 29MPa & 26MPa in bracket and ribs respectively. The variation of von-Mises stresses induced in the spar assembly w.r.t lamina stacking sequence.

From the graph it is clear that, the stacking sequence LY11/4 is having more von-Mises stress of 323MPa in spar assembly and Stacking sequence LY5/3 is having minimum von-Mises stress of 302MPa.

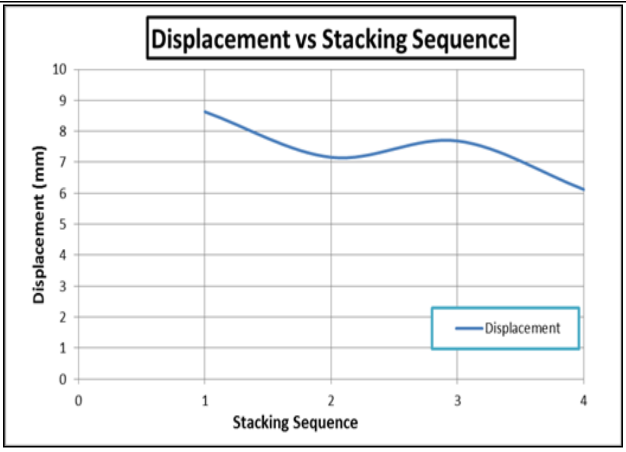

Fig 21: Displacement vs Stacking Sequencing.

From the graph it is observed that, the stacking sequence LY3/1 is having more displacement of 8.63 mm and Stacking sequence LY11/4 is having minimum displacement of 6.12mm.

From FE results it clear that, stacking sequence LY11/4 gives the better performance compare to other stacking sequences.

5. CONCLUSION

The following conclusion was made from the FE results.

- The maximum displacement of 8.63mm is observed for stacking sequence of LY 3/1- [0/35/0].
- The maximum normal stress of 318.79MPa is observed for stacking sequence of LY5/2 [0/-35/0/35/0].
- The von-Mises stress induced in the spar is less than the allowable yield limit of the material.
- From the results it is clear that, lamina stacking sequence affects the performance of the wing structure.
- From the results it is observed that, the use of lamina with 350 yields to poor performance compare to lamina with 900.
- The use of combination of 35/45/90 gives the better performance as observed in results of LY11/4.
- The displacement induced in the wing structure for stacking sequence LY 11/4 is 6.12mm and reduced by ~20% compare to other stacking sequence.
- The normal stress induced in the wing skin for stacking sequence LY11/4 is 208.10MPa and reduced by ~ 25% compare to other stacking sequence.
- The von-Mises stress induced in the brackets and ribs are almost same for all stacking sequences.
- The von-Mises stress induced in the spar for LY11/4 is 327.93MPa and it is increased by ~ 8% compare to other stacking sequences.
- From results it is clear that, stacking sequence LY11/4 can be used for the wing structure.

ACKNOWLEDGEMENTS

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REFERENCES

1. Graeme J. Kennedy and Joaquim R. R. A. Martinsy, "*A Comparison of Metallic and Composite Aircraft Wings Using Aerostructural Design Optimization*", University of Toronto Institute for Aerospace Studies, Toronto, ON, Canada.
2. Kong, h. park, y. Kim and k. Kang, "*Structural design on wing of a small scale wig vehicle with carbon/epoxy and foam sandwich composite structure*", 16th international conference on composite materials.
3. F. H. Darwisha, G. M. Atmeh and Z. F. Hasan, "*Design analysis and modeling of general aviation aircraft*", Jordan Journal of Mechanical and Industrial Engineering Volume 6, Number 2, pp 183-191, 2012.
4. M.pavithra, V.srilekha, "*static and dynamic analysis of a typical aircraft wing structure using MSC Nastran*", international journal of research in aeronautical and mechanical engineering volume 3, issue 8, ISSN: 2321-3051.
5. Jing Wang, "*Measurement of the mechanical properties of angle ply laminates*", University of Toronto Institute of Aerospace Studies, Toronto, ON, Canada.
6. Dianzi Liu, and Vassili V. Toropov, "optimization composite wing panels using smeared stiffness technique and lamination parameters", University of Leeds, LS2, 9JT, UK.
7. Levent Ünlusoy, "Structural design and analysis of the mission adaptive wings of an unmanned aerial vehicle", A thesis submitted to Middle East Technical University, February 2010.