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Analysis of Effect of Lamina Orientation on Static and Dynamic Behavior of Aircraft Wing

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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. 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 and dynamic 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 *– Static, Dynamic, Analysis, Lamina Orientation, Thickness, Strength-Weight Ratio.*

1. INTRODUCTION

The arrangement and examination of the wings of plane is one of the key usages of the investigation of streamlined components, which is a branch of fluid mechanics. The properties of the wind current around any moving article can - on a fundamental level - are found by comprehension the Navier-Stokes states of fluid components. Regardless, except for direct geometries these conditions are broadly difficult to grasp. Luckily, less difficult clarifications can be portrayed. For a wing to deliver "lift", it must be situated at a reasonable approach with respect to the stream of air past the wing. When this happens the wing avoids the wind current downwards, "turning" the air as it passes the wing. Since the wing applies a power reporting in real time to alter its course, the air must apply a power on the wing; break even with in size however inverse in bearing. This power shows itself as contrasting pneumatic forces at various surfaces on the surface of the wing A locale of low weight is framed on the upper surface of the wing while a district of high pneumatic force is shaped on the lower surface of the wing. Henceforth, a net upward drive follows up on the wing. This power is known as the "lift" created by the wing. The distinctive speeds of the air going by the wing, the gaseous tension contrasts, the adjustment in bearing of the wind stream, and the lift on the wing are naturally one marvel.

It is, in this manner, conceivable to ascertain lift from any of the other three. For instance, the lift can be computed from the weight contrasts, or from various speeds of the air above and underneath the wing, or from the aggregate energy change of the diverted air. Liquid progression offers different ways to deal with tackling these issues—and all create the same answers if done accurately. Given a specific wing and its speed through the air, wrangles over which scientific methodology is the most helpful to utilize can be mixed up by learners as contrasts of assessment about the fundamental standards of flight.

2. GEOMETRY AND FE MODELLING

This section describes about the geometry creation and assembly.

2.1 Geometry Modeling

Modeling of the aircraft wing assembly is carried out in the modeling software called SolidWorks. Individual components are created and are assembled as shown in below fig.

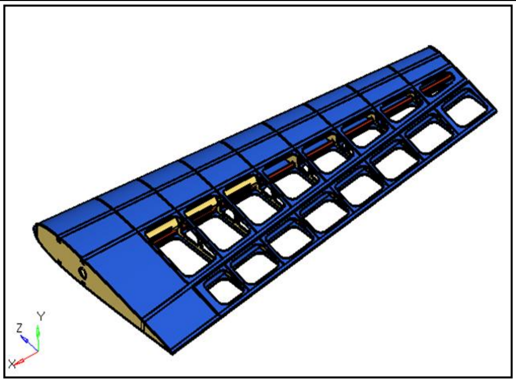

Fig 1: Aircraft Wing Assembly.

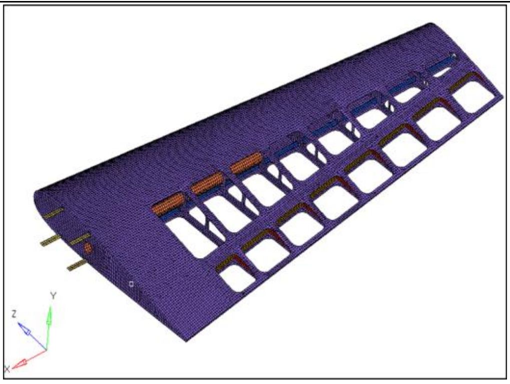

Fig 2: FE Model of Aircraft Wing.

2.2 FE Modeling

The mid surface is extracted for the assembly of the aircraft wing and is discretized with shell elements (i.e. Shell181 as per Ansys terminology) with even division of elements with size of 15mm and min of 2 rows of elements are formed for superior precision of the results. The complete FE meshed model contains 80125 number of elements and 94215 number of nodes. The above Fig 2 shows the FE model of the Aircraft wing assembly.

2.2 Boundary Condition

Boundary is one of the critical inputs to any FE analysis, as wrong Boundary Conditions leads to erroneous which cannot be predicted in the future. Hence, proper and realistic Boundary Condition needs to be assigned in the analysis. Below fig 3, shows the Boundary Conditions used in this analysis.

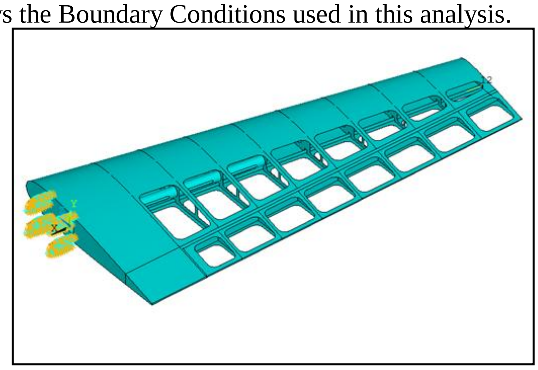

Fig 3: FE Model with Constraints.

The end of the wing is connected to fuselage. Hence, these ends are constrained for all 6 DOFs as shown in the above fig. The gravity load of 13906 mm/s^2 is applied in Y direction on complete wing assembly and the Lift load of 0.098MPa is applied on wing skin.

2.3 Material Used

Carbon Fiber - Epoxy Composites		
Properties	**Values**	**Unit**
Young's Modulus along fiber direction1 (E11)	**99170**	**MPa**
Young's Modulus along matrix direction2 (E22)	**58210**	**MPa**
Young's Modulus along matric direction3 (E33)	**58210**	**MPa**
Poisson's ratio (v12)	**0.08**	
Poisson's ratio (v23)	**0.08**	
Poisson's ratio (v13)	**0.08**	
Shear modulus in 1-2 plane (G12)	**4180**	**MPa**
Shear modulus in 2-3 plane (G23)	**4180**	**MPa**
Shear modulus in 1-3 plane (G13)	**4180**	**MPa**
Density	**1500**	**Kg/m^3**

Table 1: Material Properties.

3. LOAD CALCULATION

Aircraft under its service conditions, two major types of loads are acting on the exterior surface of the wing structure. Detailed calculation is shown below.

The lift load and gravity loads are considered as important load data for aircraft wing design.

The gravity load is given as

Gravity (g) = 9810 mm/s^2

Load Factor (f) = 1.35 (35% of the load is considered as exceptional load)

Density factor to the material (d)=1.05

Gravity = g * f * d

= 13906 mm/sec^2

The fuselage and wing are the two major components where lift load acts. It is general consideration that 80% lift load acts on wing and 20% load acts on fuselage. Hence maximum load acts at wing roots.

The lift loads are calculated from Basic aerodynamics,

$L = n * W$

Where,

L = Lift produced by the entire aircraft

n= Load factor

W = Weight of the aircraft

Since we are concentrating on structural parameters during take-off and climbing phase, lift force must be greater than the weight of the aircraft.

Weight of the aircraft = 14000N

Design Load factor = 6g

Factor of safety = 1.5

Total Design load acting on aircraft wing is,

$L = n* w$

$= 1.5*6*140000$

L = 126000 N

As discussed, 80% load acts on wing structure, hence

Total Load on wing structure = 126000*0.8

= 100800 N.

Total Load acting on each wing = 50400 N.

For analysis, this force is converted into pressure load and applied.

Pressure load = Force/ Projected area of wing

P = 50400/ 515772.3

P = 0.098 MPa.

4. RESULTS AND DISCUSSIONS

Results have been conducted for different orientation and stacking sequence keeping same boundary conditions and loads. Loads are calculated based on the empirical formula that are used to calculate wind pressure on the wings, from the equation calculated load is 130906mm/s^2.

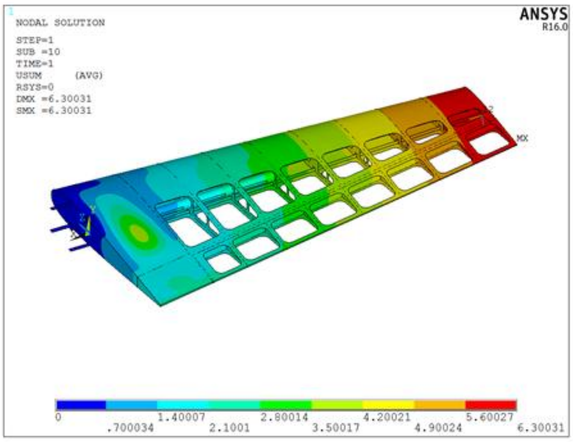

Fig 4: Maximum Displacement Plot

Maximum displacement found in the Wing model is 6.30mm and maximum X Directional stresses found in the top and bottom layer is 81.43 and 108.51Mpa. Maximum Y Directional stresses found in the top and bottom layer is 27.81 and 10.35Mpa.

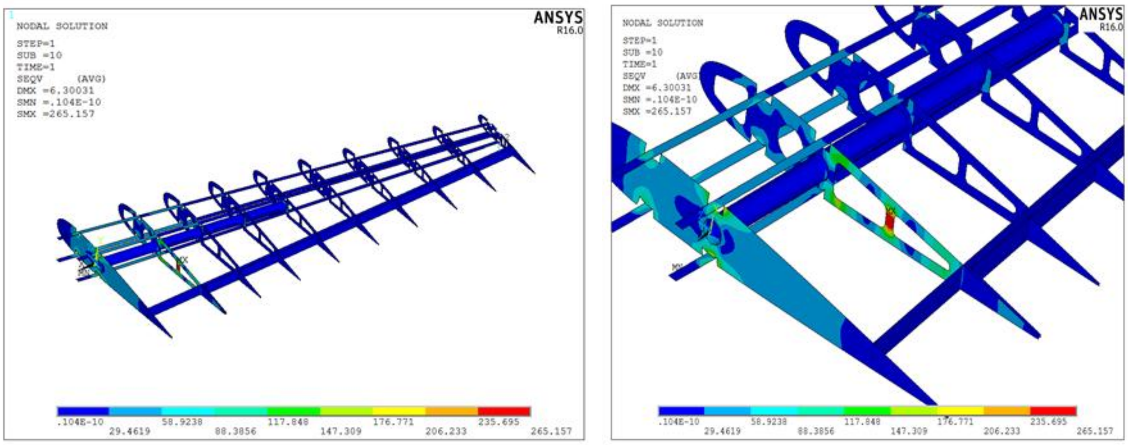

Fig 5: Maximum von-Mises Stresses in Structure and Zoom in View.

From the above fig 5, the maximum von-Mises stresses found in the model is observed to be 265.15MPa, taking this iteration as an example we have to optimum best stacking sequence ply of the material and reduced the stresses in the wing structure.

Similarly, different iterations are carried out for different lamina orientation such as (layer- 90/45/90), layer-35/-35/0/35/-35), (layer- 0/90/0/90/0).

5. CONCLUSION

From the above FE Analysis fallowing conclusions can be drawn.

we found that he maxi displacement of 6.37mm is observed in iteration LD2 [i.e. layer definition of 90/45/90] and maxi normal stress of 120MPa is observed in iteration LD4 [i.e. layer definition of 0/90/0/90/0]. The von-Mises stress induced in the spar is less than the allowable yield limit of the material and from the results it is clear that, lamina stacking sequence affects the performance of the wing structure. From the results it is clear that, the use of lamina with combination of 45^0 & 90^0 gives better results compared to lamina with 90^0 and35^0. The normal stress induced in the wing skin for layer definition LD2 is 64MPa and reduced by ~ 46% compare to other stacking sequence. The von-Mises stress induced in the spar assembly for layer definition LD2 is 216MPa and reduced by ~ 22% compare to other layer definition. From results it is clear that, layer definition LD2 can be used for the wing structure.

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