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Finite Element Analysis of Aircraft Wing Using Composite Structure

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Abstract
The thesis deals with bending Finite Element Analysis of monocoque laminated composite aircraft (subsonic and supersonic) wing using commercial software ANSYS. Theoretical background, mathematical formulation and finite element solution for a laminated composite shell structure are presented in this study. A monocoque aircraft wing is made of laminated composite with fiber angles in each ply aligned in different direction. Various airfoil thickness and ply angles were considered to study the effect of bending-torsion decoupling.

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I. INTRODUCTION

A aircraft wing is made of laminated composite with fiber. Various airfoil thickness and were considered to study the effect of bending-torsion decoupling. Results obtained are presented and parametric studies are made to show the effect of airfoil thickness variation. To reduce this aeroelastic effect it is usually solved at the level of due to aerodynamic load vibration response. The proposed solution for this aeroelastic effect in this research it is planned to decouple the bending-torsion of the airfoil in the Eigen modes. It may be noted that the dynamic response for any dynamic load is a sum of modal responses. Hence if we reduce or decouple the bending torsion of the wing for the first ten mode, it is likely that the bending-torsion coupling out of the aerodynamic force will be a minimum and the control problem.Aerospace research center will be used in this study for generating the FE model of NACA 4412 airfoil geometry for an aircraft wing using a material of composite structure. Airplane wing is often improved for better performance such as increases in flutter speed and reduction in control problem. This research will help to improve the performance of the aircraft.

II. Airfoil Theory Terminology And Definitions

2.1 Airfoil Geometry Naming Conventions

Airfoil geometry can be characterized by the coordinates of the upper and lower surface. It is often summarized by a few parameters such as: maximum thickness, maximum camber, position of maximum thickness, position of maximum camber, and nose radius.

2.2 NACA 4-Digit Series

Consider the airfoil NACA 4412. The first digit gives maximum camber in percentage of chord, the second digit gives in tenth of a chord where the maximum camber occurs, and the last two digits give the maximum thickness in percentage chord.
NACA 4 digit series:

2.3 Laminated type composite structure

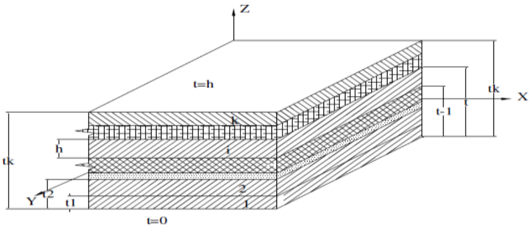
A typical composite structure consists of a system of layer bonded together. The layers can be made of different isotropic or anisotropic materials, and have different structure, thickness, and mechanical properties. The laminate characteristics are usually calculated using the number of layer, stacking sequence, geometric and mechanical properties. A finite number of layers can be combined to form so many laminates, the laminates characterized with 21 coefficients and demonstrating coupling effect. The behavior of laminates as a system of layer with given properties. The only restriction that is imposed on the laminate as an element of composite structure concerns its total thickness which is assumed to be much smaller than the other dimensions of the structure.

2.4 Stiffness Coefficient of Laminated Layer

A laminate consisting of a number of layers with different thickness h_i and stiffness $A^i_{mn} i$= 1,2,3,…..k. Assuming that material stiffness coefficient do not change within the thickness of the layer and using piece-wise integration, it is possible to write the parameter I_{mn} as:

$$I^i_{mn} = \frac{1}{r+1}\sum_{i=1}^{k} A^i_{mn}\left(t_i^{i+1} - t_{i-1}^{i+1}\right)$$

Where r=0,1,2 and t_o= 0,t_h = for thin layer



Structure of the laminate

The member, coupling and bending stiffness coefficient of the laminate are given respectively

$$B_{mn} = \int A_{mn}\, dz$$

$$D_{mn} = \int A_{mn} z^2 dz$$

$$C_{mn} = \int A_{mn} z dz$$

Where B_{mn} is member stiffness
D_{mn} is coupling stiffness
C_{mn} is bending stiffness
In plane strains of the layer ε_x,ε_y *and* γ_{xy} can be found

$$\varepsilon_x = \frac{\partial u_x}{\partial x} = \varepsilon^o_x + zk_x,$$

$$\varepsilon_y = \frac{\partial u_y}{\partial y} = \varepsilon^o_y + zk_y,$$

$$\gamma_{xy} = \frac{\partial u_x}{\partial y} + \frac{\partial u_y}{\partial x} = \gamma^o_{xy} + zk_{xy}$$

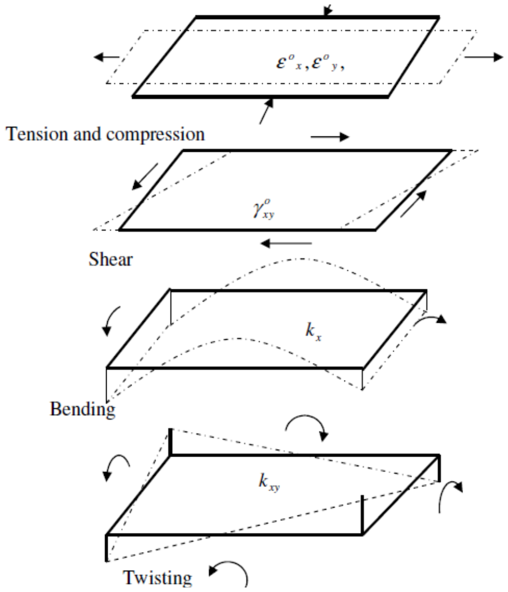
Where,

$$\varepsilon^o_x = \frac{\partial u}{\partial x}, \quad \varepsilon^o_y = \frac{\partial v}{\partial y}, \quad \gamma^o_{yx} = \frac{\partial u}{\partial y} + \frac{\partial v}{\partial x},$$

$$k_x = \frac{\partial \theta_x}{\partial x}, \ k_y = \frac{\partial \theta_y}{\partial y}, k_{xy} = \frac{\partial \theta_x}{\partial y} + \frac{\partial \theta_y}{\partial x}$$

This generalized strains corresponding to the following basic deformation of the layer shown in the figure

- In plane tension or compression(ε^0_x,ε^0_y)
- In plane shear(γ^0_{xy})
- Bending in xz-and yz plane(k_x,k_y)
- Twisting(k_{xy})



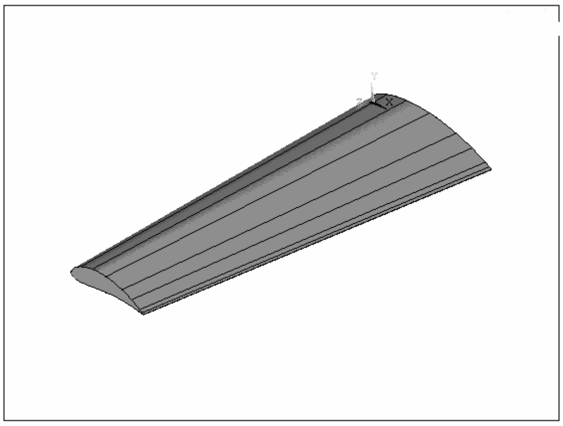
Basic deformation of the layer

III. Subsonic Aircraft Wing Model Description And Case Studies

3.1Physical Model of subsonic Aircraft wing:

The physical structure modeled in this work is a shell aircraft wing of airfoil cross section NACA 4412 series with fiber laminated composite structure, shown in Figure (3.1). Its dimensions are that of a research subsonic aircraft wing. The chord length at the free end is 0.8m and at the fixed end is 1.8m while the length of the wing is 15m. The thickness of the shell wing for some spatial distance is treated as to reduce the twist angle parameter.

Figure 3.1 Physical model of subsonic aircraft wing



The dimension of this model is a tapered aircraft wing. It is made of a current; carbon- epoxy laminated composite structure without any ribs and long irons

IV. Subsonic Aircraft Wing Model Description

4.1Physical Model of subsonic Aircraft wing:

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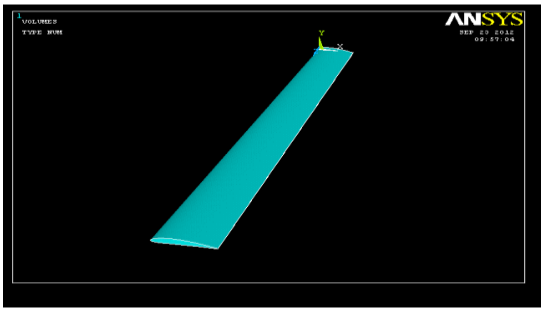

Figure 4.1 Physical model of subsonic aircraft wing

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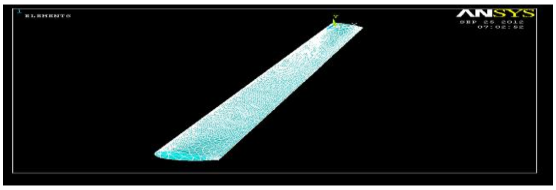

Figure 4.2 Finite element model of subsonic aircraft wing

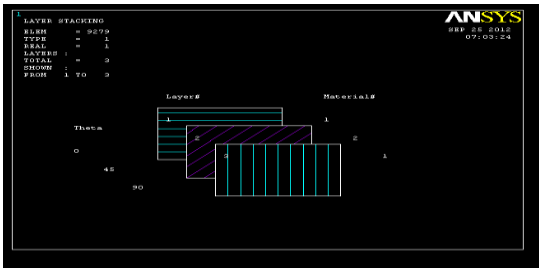

Figure 4.3 Layer stacking sequence of laminated Composite structure (subsonic wing)

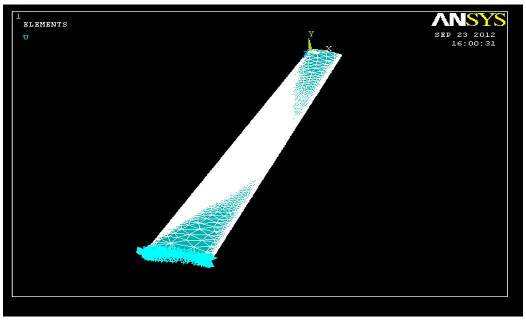

Figure 4.4 Finite element model applying boundary conditions

4.2Finite Element Model of subsonic aircraft wing

Analyses are performed in this study by using a finite element model of the aircraft wing. The model was developed in ANSYS 10.0; it has 47210 element, 74422 nodes and 3 layers. Each thickness of the layer in different spatial locations are treated as to reduce twist angle parameter. A typical finite element mesh is shown in Figure (4.2),and the layer stacking sequence are shown in figure (4.3). The global z coordinate is directed along the axis of the wing, while the global x coordinate is directed along the chord and the global y direction is perpendicular to both.

Boundary Condition

The wing is treated as cantilevered shell. That is fixed at one end (i.e. all DOF) and free at the other end, as shown in the figure(4.4)

Material Properties:

The material properties used throughout this study are shown below. These properties are for a carbon/epoxy and material Aluminium Alloy are:

1.Material Properties (Aluminium Alloy)

- Material used= Aluminium Alloy
- Young's Modulus = 73 Gpa
- Poission's Ratio = 0.3

2.Material Properties (Carbon- epoxy)

- Material used= Carbon- epoxy
- Young's Modulus = 140 Gpa
- Poission's Ratio = 0.4

V. Results And Discussions

Various results presented from Figure. 5.1 ~ 5.6 shows the structural characteristics of the aircraft wing. Figure. 2 shows the wing subjected to the self load due to gravity, the deflection is shown. Figure 6 & 7 shows the deflection due to the load applied in axial and vertical directions respectively. In Figure. 8, deflection due to both the axial and vertical loads simultaneously. The deflection due to the single loading in the axial direction shows more deflection than the vertical direction, this is because of the moment of inertia in those respective directions. However the studies should be extended to more vulnerable loads and moments too.

5.1 Wing with self load or acceleration due to gravity

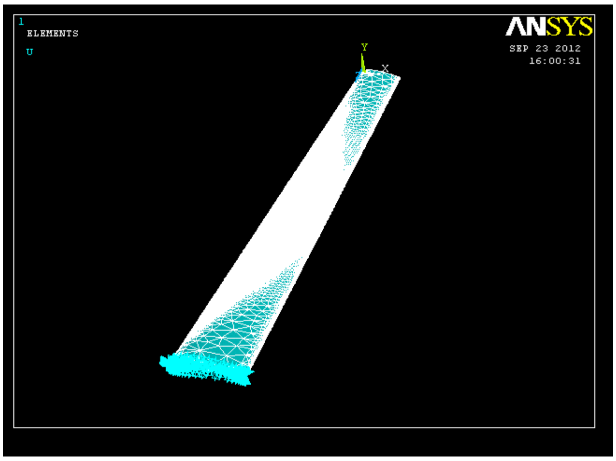

Figure. 5.1 Model fixed at the base (assumption is blade is attached to the hub)

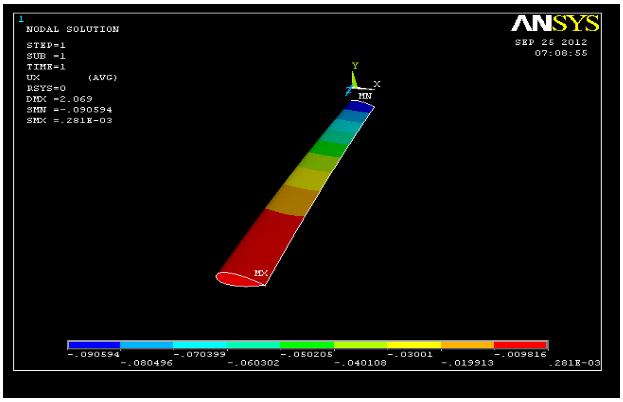

Figure. 5.2 Gravity load of 9.8 m/s^2 is applied at the wing tip.

5.2 Loading due to along wing force

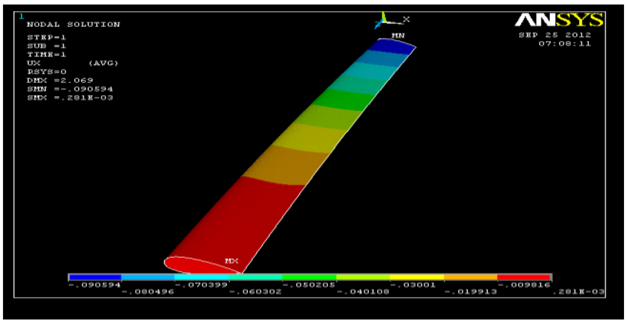

Figure. 5.3 Load acting on x-axis

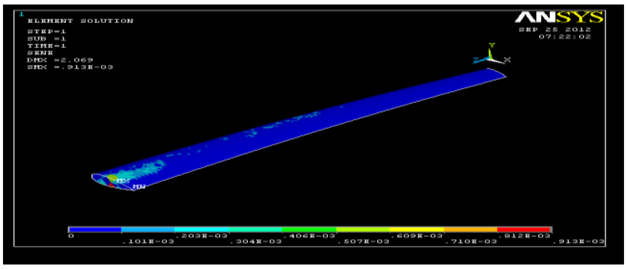

Figure. 5.4 Nodal solution for Strain energy

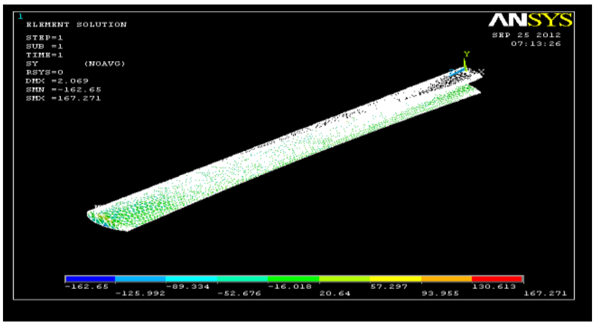

Figure. 5.5 Load acting on y-axis

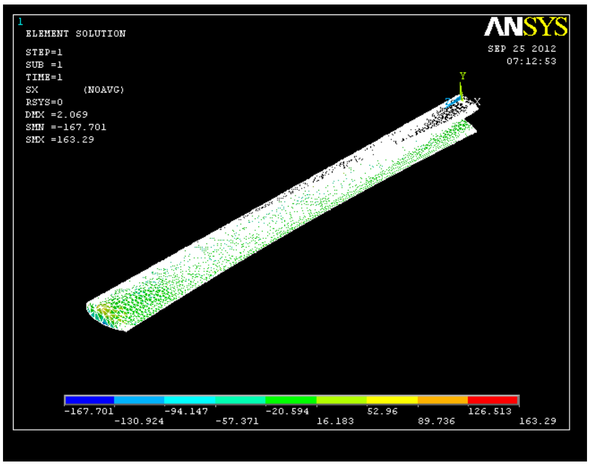

Figure. 5.6 Nodal solution for stress on x-axis

VI. Conclusion And Future Work

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. An individual loading for example the load only on X direction and its deflection in X, Y and Z directions, also the stress acting on X, Y, and Z directions are found. Von misses stress is calculated in order to know the maximum stress levels and minimum stress levels on the wing. The above mentioned results are found for the combined loading also. Their differences are shown clearly with the contour deflections, stress levels. The deflection and stress levels are shown from minimum to maximum in the color contours. Their values are given side by side. The comparisons made for the loads applied individually as well as combined loads shows the difference in values of deflection and stress levels. This model can be considered with twist for the various aerofoil shapes in future. For example NACA 4415 aerofoil or the aerofoil with different thickness can be considered.

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