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Numerical Analysis of a CFRC Aircraft Delta Wing



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ABSTRACT:

Delta wing is a aircraft wing with triangular planform used of supersonic and hypersonic aircraft. Supersonic and Hypersonic aircraft is used for high performance at high speeds. The wing in these aircrafts is used to determine the performance where it is made of steel, aluminium and its alloys, titanium and its alloys and most recent carbon fiber reinforced composites (CFRC). CFRC are the material that are made of glass fibers and carbon fibers which are set in a matrix or plastic by use of epoxy resins that are light in weight and have high strength and structural stiffness. In the present work a Delta wing of conventional NACA 4-digit airfoil is analyzed considering isotropic and composite materials in laminates. Each laminate used in this work has different ply orientations which vary with each other. A parametric study is conducted on these laminated wings using ANSYS finite element package for different orientations of composite. From these conducted ANSYS studies it is estimated the replacement of Al alloy by the CFRP's are more reliable and less in weight.

Keywords: Delta wing, CFRP, ply orientation, FEM.

INTRODUCTION:

A wing is a cross-section geometry designed to give the aircraft a maximum lift at all subsonic and supersonic speeds. A wing with sharp leading and trailing edge is desired for lift in all the supersonic aircrafts. In the current study a desired NACA 0006 airfoil having 0-percent camber at 00 of the chord from the leading edge and is 6 percent thick is considered.

Configuration of Hypersonic Aircraft Delta Wing:

The aircraft wing considered in this study is in a Delta planform having sharp leading and trailing edges having a sweep back angle of 45 degrees.

The Delta wing has a root of chord length 3000mm and a tip of chord length 50mm having a wing span of 3000mm [10]. The components of Delta wing include of NACA 4-digit airfoil of 0006 configuration having awing structure comprising of

1.Panels – Top and Bottom skin
2.Span – Front and Rear
3.Ribs – Root rib, Tip rib

The aircraft wing is subjected to the air loads acting on the surfaces of the wing directly. The wing is fixed or attached to the fuselage by the root and has a tip free to the loads acting. These loads are subjected all over the span of Delta wing where the bending of wing is happening over the root and tip with respect to the span of wing. The skin of wing is a impermeable surface that is covering the spars and ribs over the top and bottom surfaces where the root and tip are subjected to these loads. The skin of wing provides the aerodynamic pressure distribution over the wing which derives the lift capability of the wing.The pressure distribution takes place all over the wing from the root to tip over the skin of wing which has two parts i.e. the top and bottom. The primary objective of the current study is to develop a aircraft wing that can safely accomplish the flights at a minimum cost. Carbon fibers are the composites consisting of strong fibers set in a matrix of epoxy resin to form a laminate. The laminates are used as wing panels ortho-tropically or iso-tropically with other alloys of metals. The main purpose of them is to reduce the overall aircraft weight and also the production costs increased due to alloys. The CFRP has made this a possibility in recent years by advancement of composite structures.

WING MODEL:

The Delta wing having top and bottom panels is a conventional composite laminates (CFRP) with root and tip of sharp edges having the panels modeled as orthotropic CFRP laminates.

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The material properties of orthotropic materials are given in table 1

Material property	CFRP(M55j/915 prepreg
Mode of elasticity	270GPa
Mass Density	1760 kg/m3
Poisson's ratio	0.365
Tensile strength	1.8GPa
Shear strength	0.092GPa

The modeling is done in ANSYS mechanical APDL software where the composite material are modeled using specialized shell lay up elements of 3D 4node shell 181 element. The elastic shell element is used to model the intermediate components of the wing when the laminate thickness is for 6 layers of each 0.001m for various different orientations.The primary load on the wing is applied as a pressure over the surface of wing model for a mesh of finite element is shown in fig 1.The wing has configuration over its external surface consisting of top and bottom panels are made of carbon fiber reinforced polymer (CFRP) having of low coefficient of thermal expansion.

The panels are laminates of various distinct fiber orientations varying from adjoining laminates. In this study the forces is on the effects of various ply orientations on strength of panels using shell element of 3D 4 node shell 181 modeled for 6 pies each for the thickness of 1mm. The model are for 7 different wing layup orientations for angle varied from 0 to 90 degrees at 15 degree internal for each layup. The ply sequences selection in this study are tabulated in table 2

Ply sequences
$[0_2/90_2/+0/-0/90_2/0_2]$ ns
[02/902/+15/-15/902/02] ns
[02/902/+30/-30/902/02] ns
[02/902/+45/-45/902/02] ns
[02/902/+60/-60/902/02] ns
[02/902/+75/-75/902/02] ns
[02/902/+90/-90/902/02] ns

The pressure loads are applied to each wing ayup and the resultant stresses for von-misses and the displacement in each wing are studied for corresponding panels.

Ply Layout Sequence	Von-Mises Stress (N/mm2)	Displacement (mm)
$[0_2/90_2/+0/-0/90_2/0_2]$	129.414 E+02	14.3
$[0_2/90_2/+15/-15/90_2/0_2]$	277.180 E+02	19.4
$[0_2/90_2/+30/-30/90_2/0_2]$	278.231 E+02	19.3
$[0_2/90_2/+45/-45/90_2/0_2]$	280.167 E+02	19.3
$[0_2/90_2/+60/-60/90_2/0_2]$	280.481 E+02	19.3
$[0_2/90_2/+75/-75/90_2/0_2]$	293.813 E+02	17.4
$[0_2/90_2/+90/-90/90_2/0_2]$	281.113 E+02	19.5

The von-misses stress for ply layout sequences is shown in fig The resultant displacement and von-misses stresses are tabulated in table .

RESULTS AND DISCUSSION:

The aircraft wing subjected to static analysis is carried out in ANSYS software. The maximum value ofThe parametric study shows that, in the bottom plate thestress pattern for [02/902/+0/ -0/902/02] ns seems to have a more uniform distribution when compared to the other ply sequences studied.

1.The Von Mises stress value for the ply sequence [02/902/+0/ -0/902/02] ns is seen to have the least value of 129.414E+02 N/mm2.
2. The displacement corresponding to the ply sequence [02/902/+0/ -0/902/02] ns is seen to have the least value of 14.3 mm, which proves that this ply sequence is seen to have better performance.

3.Thus it is desirable to adopt the ply sequence [02/902/+0/ -0/902/02] ns for composite aircraft wings in comparison with the other ply sequences considered in the present study.

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