DESIGN AND ANALYSIS OF COMPOSITE WING FOR HYPERSONIC MISSILE

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ABSTRACT

The essence of this project is to study and analyze the effect of a wingtip missile (external store) to a supersonic fighter aircraft and to compare the result with the same configuration but without the wing tip missile. Almost all the fighter aircrafts of the modern era are already equipped with external stores. This project is divided into two categories, in the initial phase the modeling of the above said configurations is performed and in the latter part the same geometry is analyzed computationally. Upon the successfully completion both these phases it is then put forward for the comparison. For the purpose of modeling of the aircraft's geometry the required data is acquired from the technological owner of the aircraft themselves. This technical data is then elaborated to an extent which only covered the educational aspect for the purpose of the research study. A computer aided modeler namely ICEM CFD is selected for the purpose of modeling. After the completion of design process it is then analyzed numerically for the purpose of studying the effect of the external store on the aerodynamic interference on the flow around the aircraft's wing. A computational fluid dynamic (CFD) have been carried out to get the aerodynamic coefficients such as lift, drag of the wing and then certifying the aircraft behavior with and without the wingtip missile. By exact modeling and analyzing we ensure that there will not be any difficulties in stability and controllability of the wing during actual flight. CFD uses certain specific sets of equations to solve the unknown's parameters. The geometry modeled for the purpose of analysis in this project is 1/13th of the actual size. In keeping with its historic role of providing an advanced technology base in aeronautics, NASA, through its Langley Research Center, has undertaken an assessment of the significance of advanced aerodynamic, propulsion, and structural technology for cruise missile systems. Such an assessment should provide NASA with a rationale for planning a research program in those technology areas, and a nucleus of a plan for such a program. Although the assessment is centered on cruise missile systems, the recommended research programs would undoubtedly have applicability to other missile and aircraft systems. The purpose of this report is to contribute to the assessment and to recommend areas in which research and development effort is needed in aerodynamics, propulsion, and structures over the speed range from subsonic to hypersonic to support the future development of improved cruise missile systems.

INTRODUCTION

Composite materials

A composite material consists of two or more physically and/or chemically distinct, suitably arranged or distributed phases, with an interface separating them. It has characteristics that are not depicted by any of the components in isolation. Most commonly, composite materials have a bulk phase, which is continuous, called the matrix, and one dispersed, non-continuous, phase called the reinforcement, which is usually harder and stronger. The arrangement of the fiber in the matrix critically influences the composite properties.

Components in Sandwich Composites

Sandwich composites primarily have two components namely, skin and core. If an adhesive is used to bind skins with the core, the adhesive layer can also be considered as an additional component in the structure. The thickness of the adhesive layer is generally neglected because it is much smaller than the thickness of skins or the core. The properties of sandwich composites depend upon properties of the core and skins, their relative thickness and the bonding characteristics between them.

A wide variety of materials are available for use as skins. Sheets of metals like aluminum, titanium and steel and fiber reinforced plastics are some of the common examples of skin materials. In case of fiber reinforced skins, the material properties canbe controlled directionally in order to tailor the properties of the sandwich composite. Fiber reinforced polymers are used widely as skins due to their low density and high specific strength. Another advantage offered by the use of polymer composites in skins is that the same polymer can be used to make the skin and the core. Cross-linking of polymer between core and skin would provide adhesion strength level equal to the strength of the polymer. This provides possibility of making the skin an integral part of the structure eliminating the requirement of the adhesive. When an adhesive is used to bond the skin and the core together, selection of adhesives becomes very important, as they should be compatible with both the skin and the core materials. The adhesion must have desired strength level and should remain unaffected by the working environment.

In case of metallic components, welding or brazing is used as a means of binding the core and skins together. Use of adhesives is also possible but is limited to such cases where one or more of the components cannot withstand heat.

Choice of skins is important from the point of view of the work environment as this part of the structure comes in direct contact with the environment. Corrosion, heat transfer characteristics, thermal expansion characteristics, moisture absorption and other properties of the whole sandwich composite can be controlled by proper choice of skin material. In most cases both skins of the sandwich are of the same type, but could be of different type depending upon specific requirements. Difference may be in terms of materials, thickness, fiber orientation, fiber volume fraction or in any other possible form.

LITERATURE REVIEW

2.1 Characterization of synthetic foam and their Sandwich composites: modeling and experimental approach by Nikhil Gupta

The particle filled polymers known as syntactic foams are lightweight and highly damage tolerant. Syntactic foams are used as core materials in sandwich composites. The use of such materials in aeronautical and space structures make it necessary to understand their characteristics for various environmental and loading conditions. The first part of the present work takes modeling and finite element analysis approach to understand and predict the deformation behavior of syntactic foams. Contact analysis is performed on single particle models by the finite element analysis approach. In the second part extensive experiments are carried out to characterize syntactic foams for hydrothermal and compressive properties, and syntactic foam core sandwich composites for compressive and flexural properties. Flexural tests are carried out in three and four point bending and short beam shear configurations. Syntactic foams are tested in three different specimen sizes and orientations to characterize them as per the recommendations of various ASTM standards. Effect of specimen aspect ratio on the measured mechanical properties can be determined by such an approach. The effect of change in the internal radius of hollow particles, called chemosphere's, on mechanical properties is studied for all these loading conditions. Five different types of chemosphere are selected for the study of the internal radius dependence of mechanical properties of syntactic foams and their sandwich composites.

Refined model for point bending and transverse stresses of sandwich panels by V. APolyakov, I. G. Zlfigun, R. P. ShHtsa, and V. V. Klfitrov

Polyakov, I. G. Zlfigun, R. P. ShHtsa, and V. V. Klfitrov have studied about transverse stresses According to the relationships derived in [1], transverse normal and tangential stresses in a sandwich panel have been analyzed. Asymptotic formulas for the stress concentration area in the vicinity of point forces are derived. Analytical estimates of a normal stres~ at the central and end section~ of the panel are deduced. The Saint-Venant effect of the degeneration of a panel of finite length into an infinite strip is studied. For the estimation of the concentration of the transverse tangential stress, the possibility of a superposition of the solution of the slippage problem of the face layers and the classical solution allowing for shear is substantiated. It is shown that the local Reissner~type effects are specified by reducing the concentration of the tangential stress in the face layers along the longitudinal coordinate and transition to the steady tangential stress state in the filler layer. The concentration coefficients of the tangential stress are derived as functions of the dimensional parameters of the panel section.

Global-local Analysis of Laminated Composite Plates by Hashem M. Mourad, Todd O. Williams, Francis L. Addessio, T-3

Hashem and Williams has proposed technique is a finite element formulation, based on the multi scale plate theory developed by Williams, and represents the first step toward the development of an efficient computational framework that can be used to study the response and damage distribution within composite laminates subjected to dynamic impact loading.

Recent developments in finite element analysis for laminated composite plates by Y.X. Zhang , C.H. Yang

Y.X. Zhang, C.H. Yang has studied a review of the recent development of the finite element analysis for laminated composite plates from 1990. The literature review is devoted to the recently developed finite elements based on the various laminated plate theories for the free vibration and dynamics, buckling and post buckling analysis, geometric nonlinearity and large deformation analysis, and failure and damage analysis of composite laminated plates. The material nonlinearity effects and thermal effects on the buckling and post buckling analysis, the first-ply failure analysis and the failure and damage analysis were emphasized specially. The future research is summarized finally.

2.5 Determination of deflection function of a composite cantilever beam using theory of anisotropic elasticity by Alaattin Aktas

Alaattin Aktas has studied about deflection function of orthotropic cantilever beam when it is subjected to point and distributed load which are obtained using anisotropic elasticity, the deflection at the free end of the beam are calculated numerically using obtained formulas for different fiber direction.

2.6 Experimental and analytical evaluation of lateral buckling of frp composite cantilever I-beams by Pizhong Qiao and Guiping Zou, The University of Akron, Akron, OH Julio F. Davalos,

Pultruded Fiber-reinforced plastic (FRP) shapes (beams and column) are thin-walled or moderately thick-walled open or closed sections consisting of assemblies of flat panels. Due to the high strength-to-stiffness ratio of composites and thin-walled sectional geometry of FRP shapes, buckling is the most likely mode of failure before material failure for FRP shapes. In this paper, a combined analytical and experimental approach is used to characterize the lateral buckling of pultruded FRP composite cantilever I-beams. An energy method based on nonlinear plate theory is developed, and it includes shear effects and bending-twisting coupling. Three types of buckling mode shape functions (exact transcendental function, polynomial function, and half simply-supported beam function), which all satisfy the cantilever beam boundary conditions, are used to derive the critical buckling loads, and the accuracy of these approximations are studied and discussed. The effects of tip-load position, fiber orientation and fiber volume fraction on the critical buckling loads are investigated. Four common FRP I-beams with different cross-sectional geometries and various span lengths are experimentally tested, and the critical buckling loads are measured.

Fiber-reinforced plastic (FRP) structural shapes (beams and columns) have shown to provide efficient and economical applications for civil engineering construction (e.g., in bridges, piers, retaining walls, airport facilities, storage structures exposed to salts and chemicals, and others). Most FRP shapes are thin-walled structures and manufactured by the pultrusion process. The material constituents for low-cost pultruded FRP shapes commonly consist of high-strength E-glass fiber and vinylester or polyester polymer resins, and due to this choice of materials, the structures usually exhibit relatively large deformations and tend to buckle globally or locally. Consequently, buckling is the most likely mode of failure before the ultimate load reaches the material failure.

A long slender beam under bending about the strong axis may buckle by a combined twisting and lateral (sideways) bending of the cross section. This phenomenon is known as lateral buckling, and an extensive review of analytical and theoretical investigations for steel and FRP composite beams has been presented in . In this paper, a combined analytical and experimental study on lateral buckling of FRP cantilever I-beams is presented, and simplified equations for lateral buckling analysis are developed. Three different types of shape functions (exact transcendental function, polynomial function, and half simply-supported beam function), which all satisfy the cantilever beam boundary conditions, are used to obtain eigen value solutions, and their numerical results are compared and discussed. The position of applied load through the cross section at the loading tip is also considered in the formulation. Four different geometries of FRP I-beams with varying span lengths are tested under tip loadings and cantilevered restrained at the other ends. The analytical solutions are compared with finite element studies and experimental tests. Parametric studies are further performed to study the fiber angle and fiber volume fraction effects on the lateral buckling behavior.

DESIGN & ANALYSIS OF WING

Composite laminates have been used increasingly in a variety of industrial areas due to their high stiffness and strength-to-weight ratios, long fatigue life, resistance to electrochemical corrosion, and other superior material properties of composites. A true understanding of their structural behavior is required, such as the deflections, buckling loads and modal characteristics, the through thickness distributions of stresses and strains, the large deflection behavior and, of extreme importance the failure characteristics, for obtaining strong, reliable multi-layered structures. Finite element method is especially versatile and efficient for the analysis of complex structural behavior of the composite laminated structures. Using finite element method, a significant amount of research has been devoted to the analysis of buckling and post buckling, failure and damage analysis and etc.

FE analysis

The composite wing analysis with solid model is carried out using ANSYS 11. 3D layered and shell-99 elements were used to analyze and optimize the wing layup ply sequence. Various model options with tapering thickness from 3.6mm were modeled and analyzed. These options were even analyzed with varying fiber orientation to find out optimum ply sequence.

The FE model and results are given in **Fig.3.3 to 3.11.**

Option 1: Constant thickness model with 3.6 mm.

Layup: 45/-45/fiber/45/fiber/45/fiber/-45/45 where fiber angle were varied for 0, 30, 45, 60, 90.

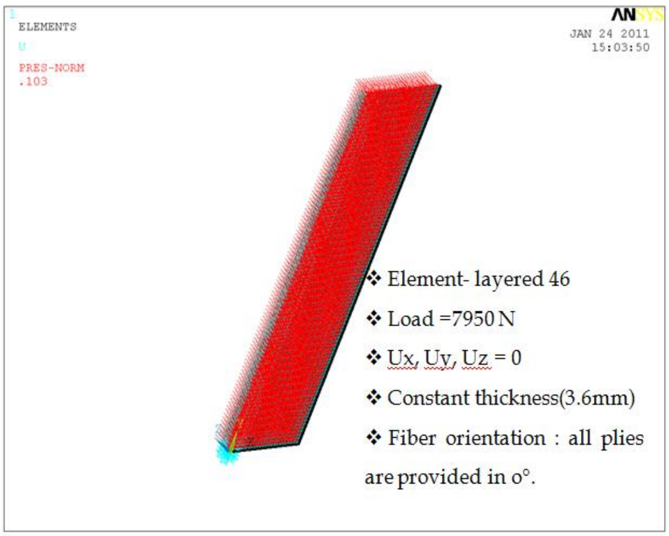

Fig.- FE model (option-1)

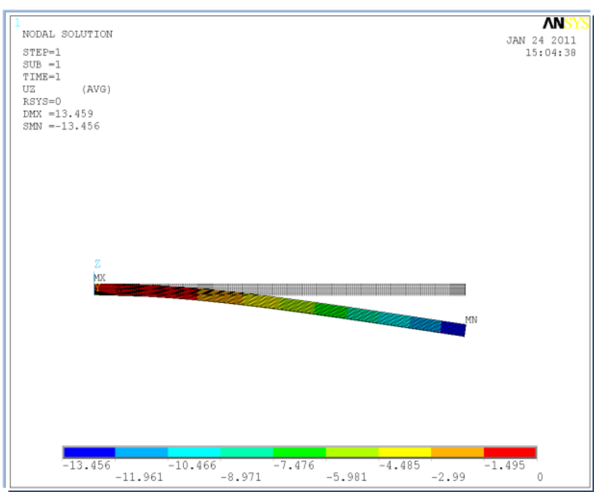

Fig.- Deflection at free end (option-1)

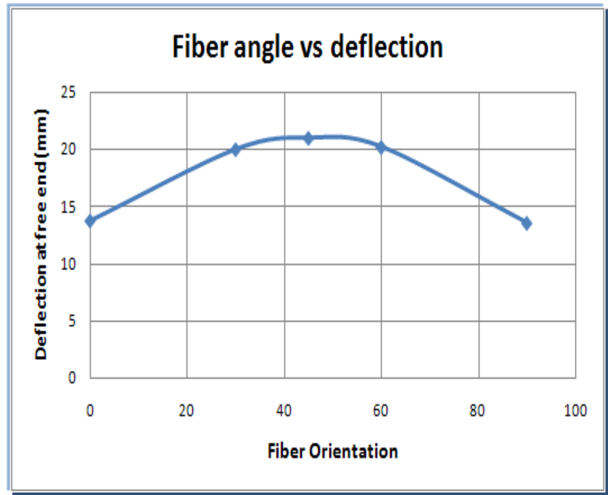

Fig. Fiber angle Vs deflection observed for option 1

Option 2: Thickness varied from 3.6 mm to 2.8 mm.

Layup: 45/-45/0/45/0/45/0/-45/45 where fiber angle were varied for 0, 30, 45, 60, 90.Consecutive layers are dropped in between to obtain the desired thickness.

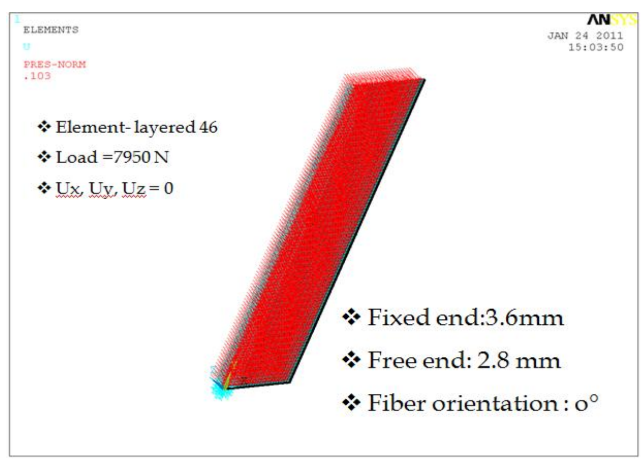

Fig.- FE model (option-2)

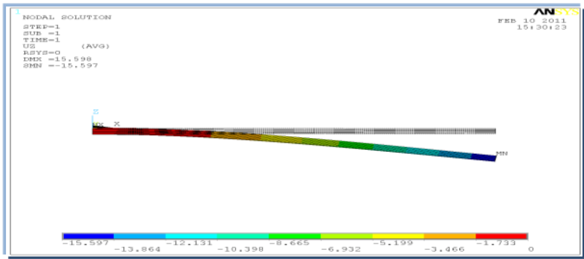

Fig. Deflection at free end (option-2)

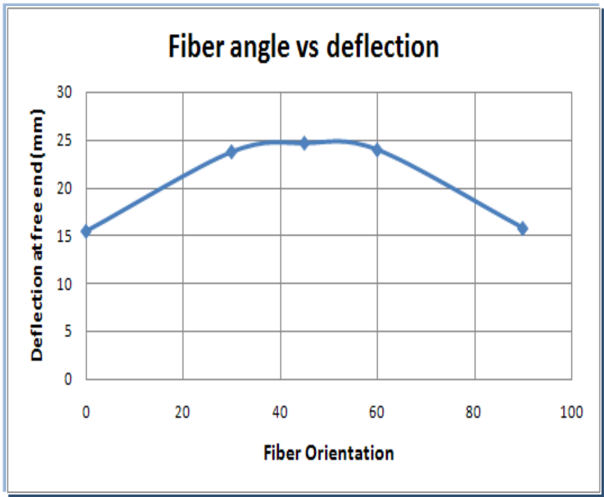

Fig.- Fiber angle Vs deflection observed for option-2

Option 3: Thickness varied from 3.6mm to 2.0 mm.

Layup: 45/-45/0/45/0/45/0/-45/45 where fiber angle were varied for 0, 30, 45, 60, 90. Four layers are dropped in between to obtain taper.

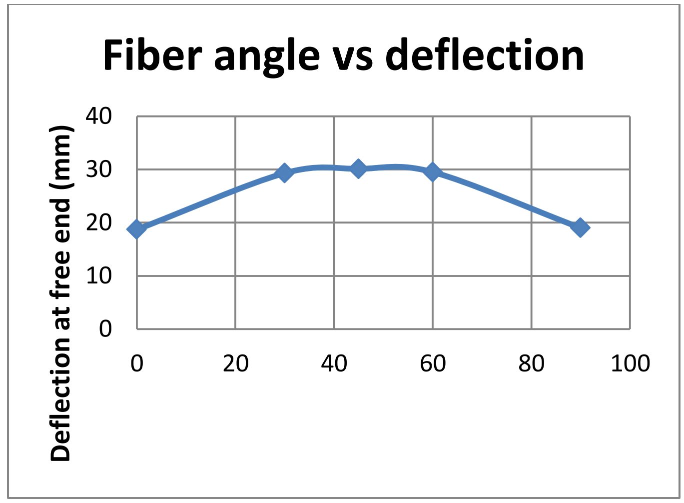

Fig.-Fiber angle Vs deflection observed for option-3

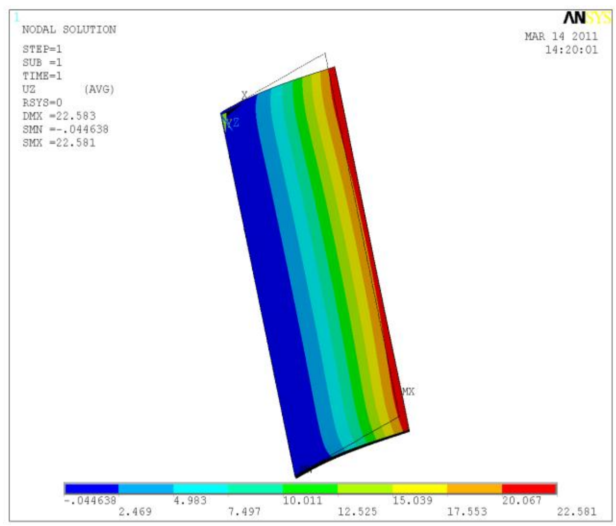

Fig.- Deflection plot for optimum ply sequence obtained in option-3

Analysis with dropping layers

The wing model of option-3 is analyzed with SHELL99 element type with layer droppings to achieve the desired model tapering from 3.6 mm to 2 mm. SHELL99 is used for layered applications of a structural shell model. While SHELL99 does not have some of the nonlinear capabilities of

SHELL91, it usually has a smaller element formulation time. SHELL99 allows up to 250 layers. If more than 250 layers are required, a user-input constitutive matrix is available. The element has six degrees of freedom at each node: translations in the nodal x, y, and z directions and rotations about the nodal x, y, and z-axes. The element is defined by eight nodes, average or corner layer thicknesses, layer material direction angles, and orthotropic material properties. Midsize nodes may not be removed from this element. The layer dropping method is shown in Fig.3.12.

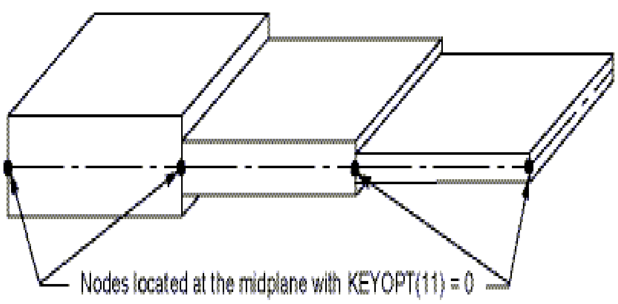

Fig. 3.12: Ply dropping available through Shell99.

SHELL99 Assumptions and Restrictions

- Zero area elements are not allowed. This occurs most often whenever the elements are not numbered properly.
- Zero thickness layers are allowed only if a zero thickness is defined at all corners. Tapering down to zero is not allowed.
- If KEYOPT(11) = 0, all nodes are assumed to be at the mid thickness of the element.(option used for analysis)
- All inertial effects are assumed to be in the nodal plane, i.e., unbalanced laminate construction and offsets have no effect on the mass properties of the element.

The optimum ply sequence obtained in option 3 is analyzed with Shell99 also. The FE model and results with this analysis are given in Fig.3.13 to 3.19.

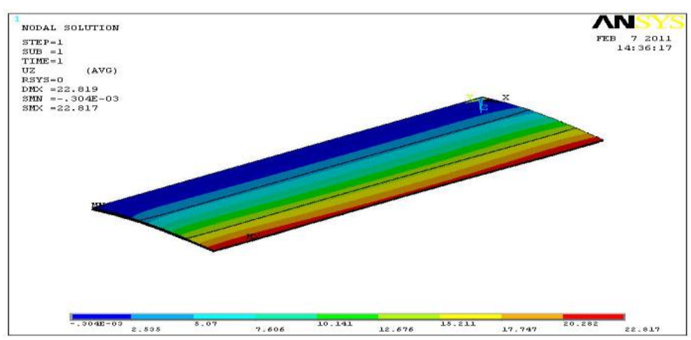

Fig. - Deflection plot

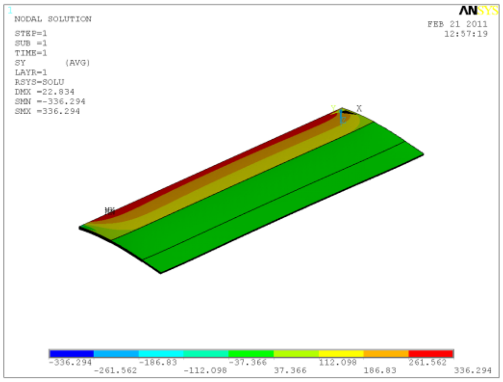

Fig. Axial Stress plot

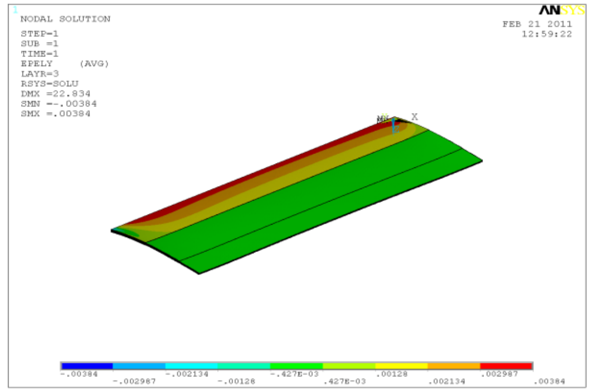

Fig.- Strain in X-direction

Table : Summary of dropping layer analysis for the two element types used

	Layered-46	Shell-99
Design load (N)	7950	7950
Displacement at free end (mm)	22.58	22.8
Axial stress (MPa)	336	336.2
Stress in -X direction (MPa)	342.1	344.0
Axial strain (micro-strains)	3651	3840
Strain in -X direction (micro-strains)	3860	3900
Factor of safety based on maximum stress.	1.6	1.6
Factor of safety based on maximum strain.	2.07	2.05
Failure load (kN)	12.72	12.72

MANUFACTURING AND TESTING

Manufacturing of mould

The most convenient way of realizing a composite wing is with a metallic mould.

The manufacturing steps followed are as follows:

1) Designed the mould using MS/EN8 material with load considered as 30T.
2) Top plate is machined from the MS block followed by taper machining using CNC milling machine.
3) Same process adopted for bottom plate as well as spacer.
4) Both the plates are chrome plated for about 3microns.This is done for easy removal of components.

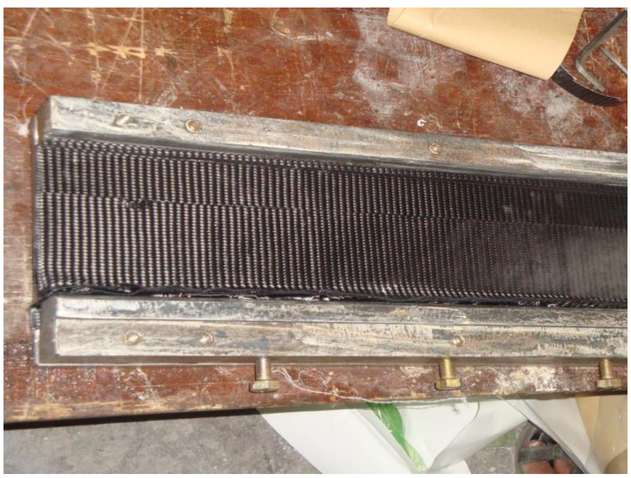

Fig: Lay-up for wing

Fig. Mould during curing

i) After curing, the mould is dismantled and the component is taken out. The component is trimmed, cleaned and resin applied to ensure moisture is not absorbed by the component. The final component is shown in Fig.4.9

Fig Composite wing

Experimental setup & load test

The composite wing under compression load tested in INSTRON TESTING MACHINE the machine is having 10 tons capacity. The strain gauges are bonded (6 No.) on the component. The locations of strain gauges are given in Fig.4.10.

1) Two gauges are bonded to wing bottom side, length from the free end is 30mm.
2) Two gauges are bonded to wing bottom side, length from the free end is 65mm.
3) Two gauges are bonded to wing bottom side, length from the free end is 120mm.

The composite wing has to undergo various tests before design and fabrication process is proved and these tests form the qualification tests. Every wing manufactured as per design has to undergo a series of tests like dimensional inspection and ultrasonic tests before it is accepted for use. These tests are called the acceptance tests. The composite wing that undergoes all the acceptance tests successfully will be made available for further use.

Figure - Strain gauge locations

Test objectives

The primary objective of the test is validation of design as well as fabrication process of the composite wing. The following are details of the objectives:

- ❖ To prove structural integrity of composite wing.
- ❖ To establish the design margins.
- ❖ To compare the predicted strains and deformations.

Functional specifications

Load = 650 kg

Ultimate load = 1.33 x Load

= 7950N

Load test procedure

The composite wing is first loaded from 0 to 0.5 kN and remove the load back to 0 thrice so that the strain gauges stabilize. The accuracy of these load cells shall be minimum 0.5 Ton. The composite wing is loaded in steps of 0.5, 1.0, 1.5, 2.0, 2.5, 3.0, 3.5, 4.0, 4.5, 5.0, 5.5, 6.0, 6.5, 7.0, 7.5 and 7.95. Strain gauge readings as well as LVDT readings, at every step, shall be recorded. Further, computer prints of strain gauge and LVDT readings shall be obtained, if possible.

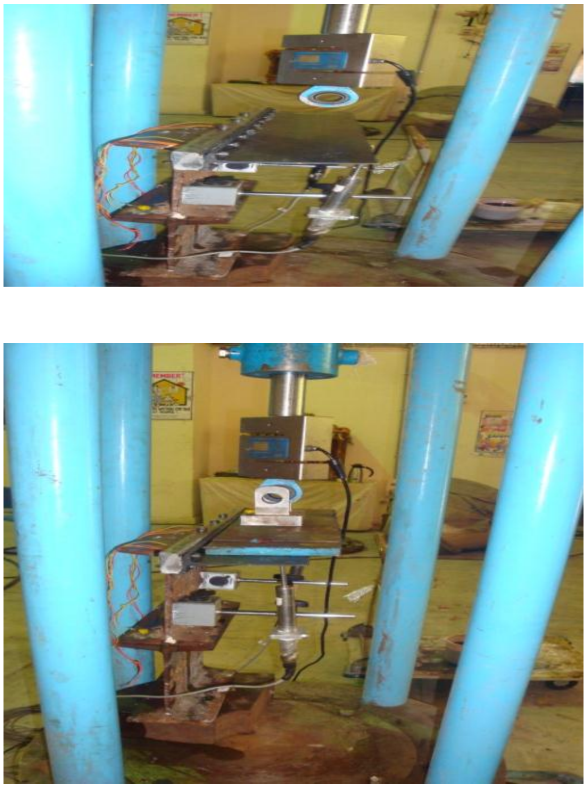

Fig.–Test setup of wing

The comparison is based on FE analysis and load test performed on the actual composite wing. The detailed test results are as follows:

Test Results

The wing of the motor case is build up by 18 load carrying CFRP layers gives a total thickness of 3.6mm. A laminate analysis was used to calculate the stress loads in the fiber reinforcement layers. An applied load of 7950 N was used in the analysis. The FE results and test results at fixed end are shown in table-5.1 and data was plotted in Fig.5.1 and Fig.5.2.

Table-5 -Axial strain & Transverse strain data (Micro-strains)

RESULTS & DISCUSSIONS

Load (kN)	Based on FE		Based on testing (micro-strains)	
	Axial	Transverse	Axial	Transverse
0	0	0	0	0
0.5	241	231	248	259
1.0	483	463	497	518
1.5	724	694	745	777
2.0	966	925.2	994	1036
2.5	1207	1156.5	1242	1296
3.0	1450	1388	1491	1555
3.5	1690	1619	1739	1814
4.0	1932	1850	1987	2073
4.5	2173	2082	2236	2332
5.0	2415	2313	2484	2591
5.5	2656	2544	2733	2850

6.0	2898	2775.5	2981	3109
6.5	3139	3007	3230	3369
7.0	3381	3239	3478	3628
7.5	3622	3679	3726	3887
7.95	3840	3900	3950	4120

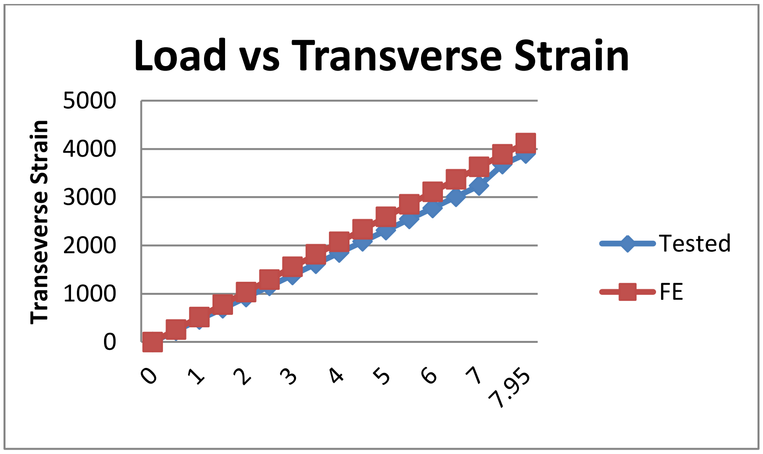

Fig.–Transverse Strain plot

Displacement at free end

The FE results and test results are given in table-5.2 and data was plotted in Fig.5.3

Table-6. Displacement details

Load (kN)	FE results	Test data
0	0.00	0.00
0.5	1.43	1.46
1	2.87	2.92
1.5	4.30	4.38
2	5.74	5.84
2.5	7.17	7.30
3	8.60	8.75
3.5	10.04	10.21
4	11.47	11.67
4.5	12.91	13.13
5	14.34	14.59
5.5	15.77	16.05
6	17.21	17.51
6.5	18.64	18.97
7	20.08	20.43
7.5	21.51	21.89

7.95	22.80	23.20

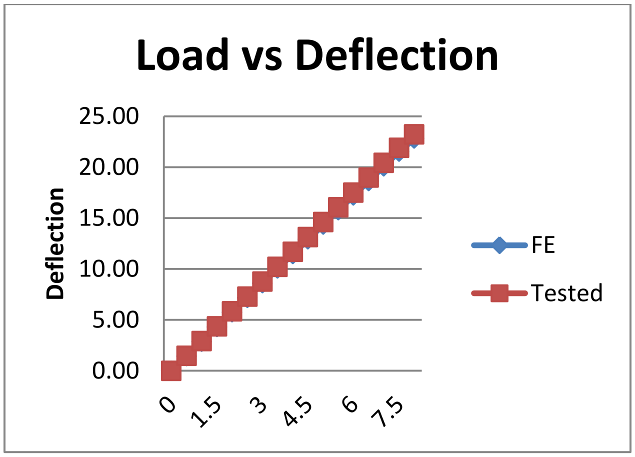

Fig. –Displacement plot

Discussion

Compared the FE results and test results based on long plate theory and it is observed that the results achieved are identical to each other

The deflection calculated based on FE analysis = 22.8 mm

The deflection measured during testing = 23.2 mm

Fixed end of wing having maximum stresses occurring because sudden change in the meridian contour. Based on testing, the estimated failure load is 15.43 kN against a failure strain in carbon fabric/ epoxy composite is 7800-8000 micro-strains. They show good agreement at all the regions. However, the result finite element analysis shows close agreement with the experimental results. Also, the design stresses were within safe limits.

CONCLUSION

The composite wing is successfully designed to meet the stipulated loading condition. The geometry of the wing has been obtained after giving it through a analytical treatment. The design approach is based on long plate theory. The wing is fabricated using carbon fibre and fabric epoxy composite by optimum ply orientation. Layup: 45/-45/fiber/45/fiber/45/fiber/-45/45. In design of composite the following configuration are considered:

- Wing with varying thickness is analyzed.
- Different composite material is considered.
- Varying ply orientations are tested.
- Layered dropping has been analyzed.

The Composite wing was analyzed by SHELL99, SOLID-45 based on analysis following conclusion are been drawn:

- The composite wing design is meeting the stipulated composite load.
- The new method proposed for analysis of wing worked out to be efficient in accurately predicting the structural response.
- The FE results and testing results are in closed agreement.
- The design stresses are within limits.

REFERENCES

1. Nikhil Gupta, Characterization of synthetic foam and their Sandwich composites: modeling and experimental approach
2. A Refined model for point bending and transverse stresses of sandwich panels by A.Polyakov, I. G. Zlfigun, R. P. ShHtsa, and V. V. Klfitrov
3. Global-local Analysis of Laminated Composite Plates by Hashem M. Mourad, Todd O. Williams, Francis L. Addessio, T-3
4. Recent developments in finite element analysis for laminated composite plates by Y.X. Zhang , C.H. Yang
5. Determination of deflection function of a composite cantilever beam using theory of anisotropic elasticity by Alaattin Aktas
6. Pizhong Qiao and Guiping Zou, The University of Akron, Akron, OH Julio F. Davalos, West Virginia University, Morgantown, WV