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Design, Fabrication, Vibrational Testing and Analysis of Composite Wingboxusing E-Glass Epoxy Composite

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Abstract:
In general sense, wing can be assumed to be cantilevered to the fuselage. All airplane wings need longitudinal members to sustain the bending moments. These moments are caused due to lift force which acts upwards. Thus the lower cover is loaded primarily in tension and upper cover is loaded primarily in compression. As a result of the all lift forces evolved, there is a large moment created at the intersection of the wing and fuselage. Those moments cannot be sustained by wing and fuselage attachments. All these moments are withstand by Wingbox which connects with to the fuselage. The present investigation deals with the manufacture and testing of the wingbox to derive its fundamental mode shapes.

Keywords:
Fuselage, wing, lift, moment, composite, modal analysis, impact hammer, wingbox, E-Glass epoxy.

I. INTRODUCTION:
The structural design of an airframe is determined by multidisciplinary criteria (stress, fatigue, buckling, control surface effectiveness, flutter and weight etc.). Several thousands of structural sizes of stringers, panels, ribs etc. have to be determined considering hundreds of thousands of requirements to find an optimum solution, i.e. a design fulfilling all requirements with a minimum weight or minimum cost respectively. The design process involves various groups of the airframe manufacturer and its suppliers, and requires the application of complex analysis procedures to show compliance with all design criteria. Traditionally the structural sizes of a wing box are determined by the stress group of the airframe manufacturer or its supplier. This is done by analysing the stress and buckling reserves for a few selected loads. Modification of the structural sizes usually affects not only local stresses but also the internal load distribution. Therefore, this approach requires an iterative, complicated and time-consuming process. Since the design process is performed with a few dominating load cases only, there is a risk of not meeting the design criteria for the complete set of design driving load cases. Furthermore, fatigue requirements are only considered on an approximate basis[2], [3].This can result in re-work and additional cost when the full set of load-cases and fatigue criteria are considered later in the design process. Due to resources and time limitations, the manual iterative process is usually stopped after achieving a design which is feasible, from a strength viewpoint, and which is close enough to the target weight. This design is not necessarily a minimum weight design. A typical schematic of a wingbox is shown in Fig.1.

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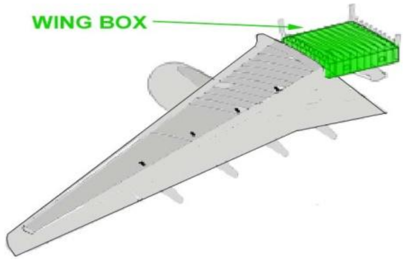

Fig.1. Wingbox

II. METHODOLOGY

As per the literature survey [1], the outer dimensions of the wingbox at root section should be as follows

Length	320 mm
Width	370 mm
Height	70mm

TABLE 1. Scaled Configuration of Wing Box

The load distribution[1] along the span of the wingbox is shown in Fig.2, Shear force and bending moment calculations for the selected wing are given in Fig.3 &4.

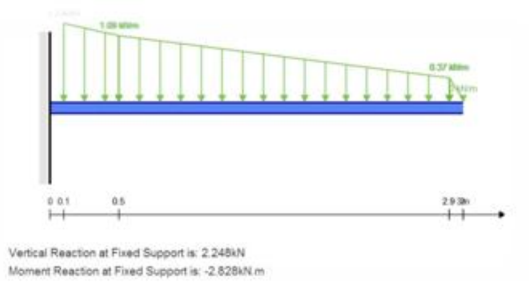

Fig. 2. Wing Loading

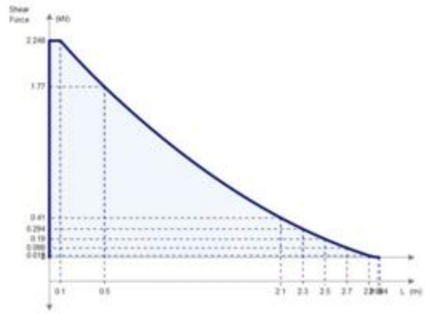
Fig.3. Shear force diagram

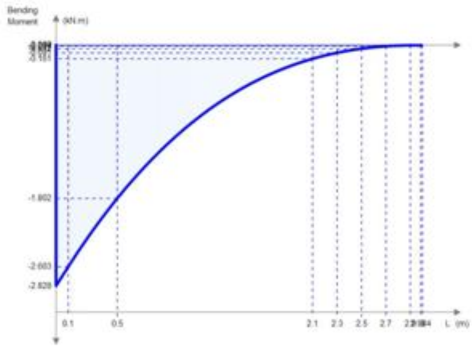
Fig.4. Bending Moment Diagram

By using deformation theories as follows, we can formulate an equation to find the thickness of wingbox C-section[5],[6,],[7], [8]

$$\frac{M}{I} = \frac{F}{Y}$$

M = Bending moment

F = Flexural strength

Considering the aerofoil section in the wing to be a box section for calculations convenience (Fig.5). The moment of inertia at each station is calculated, which is the function of t (composite thickness). We will get equation in terms of t, composite thickness is obtained after solving the equation. Span-wise wing thickness is obtained. The chord-wise thickness is obtained by CFD analysis. The aerofoil is divided into 5 zones chord-wise. With the varying pressure values in zones the thickness is obtained. The estimated composite thickness for one ply from previous results is 0.5. Therefore to get 7.5 thickness 15 plies are used.

CALCULATIONS

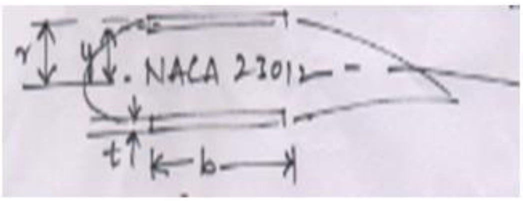

Fig.5. Showing Wing Box Dimensions

Bending moment at station 0

$$r = \frac{maximumthickness}{2} = \frac{142}{2} = 71$$

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$$I_{xx} = 2[\frac{bt^3}{12} + AY^2]$$

b is 55% of chord=653mm

t thickness of composite unknown

$$\frac{M}{I} = \frac{F}{Y} = Z$$

$$I_{xx} = 2[b \times \frac{t^3}{12} + \left[(b \times t)\left(y - \frac{t}{2}\right)^2\right]]$$

`Finally t=5.68 at station ‘0’

Considering the factor of safety and ply drop-off, the minimum thickness of composite is increased to 7mm

t=7mm

III. DESIGN OF MOULD

The Matched Die Molds(Fig.6) are initially designed in CAD software and manufactured. These moulds are used to make the required composite parts.

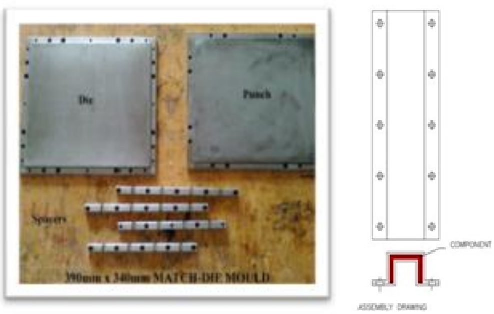

Fig.6. Matched DieMolds

IV.SELECTION OF MATERIAL

A. E-Glass Fabric

The use of E-Glass Fabric as the reinforcement material in polymer matrix composites is extremely common. Optimal strength properties are gained when straight, continuous fibers are aligned parallel in a single direction. To promote strength in other directions, laminate structures can be constructed, with continuous fibers aligned in other directions. Such structures are used in storage tanks and the like.

Technical specifications:

1. Nomenclature	:	13 mil E-GLASS FABRIC
2. Thickness, mm	:	0.36
3. Width, inch	:	40"
3. Weave	:	4 Harness Satin

B. Resin and Hardener

Resin and hardener used in this project are Lapox L-12 (Resin) and K-6 (Hardener) respectively.

V.FABRICATION OF WING BOX

As the other layup techniques involve lot of workload, equipment and costly and time consuming we preferred to use the hand layup assisted Matched Die Molding technique as it exactly suits our requirements.

A. Fabrication of E-Glass Epoxy Laminates & C-Sections Single layer of a laminated composite material is generally referred to as a ply or laminate. It usually contains a single layer of reinforcement, unidirectional or multidirectional. A single lamina is generally too thin to be directly used in any engineering application. Several laminae are bonded together to form a structure termed as laminate. Properties and orientation of the laminae in a laminate are chosen to meet the laminate design requirements. Properties of a laminate may be predicted by knowing the properties of its constituent laminae. The various steps involved in the manufacture of composite laminate are

1. Marking the fabric as per the mold dimensions
2. Mixing of matrix (Resin and Hardener (1 : 10))
3. Application of resin mix on the fabric
4. Lay up on the mold
5. Closure of Mold

Finally it is allowed for 24 hours to cure the rectangular laminate / C-section.After the curing is over, the laminate and C-Sections are trimmed using diamond edge cutter at the edges to match the planned dimensions. The specifications of the rectangular laminate and C-section are given in the following tables.

Rectangular laminate specifications

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Length	370mm
Width	320
Thickness	5mm
Number of laminates	2

TABLE 2. RECTANGULAR LAMINATE SPECIFICATIONS

C-section specifications

Length	3700mm
Width	40mm (WEB) and 70mm(flange)
Thickness	16mm

TABLE 3. C-Section Laminate Specification

Fig.7. Top view of wing box

Fig.8. Front View of Wing Box

Dimensions of wing box.

Length	370mm
Width	320mm
Height	80mm
Number of rivets	28

TABLE 4. Dimensions of Wingbox

VI. TESTING AND ANALYSIS

A. computational modal analysis

Part	Dimension
Rectangular plate	L=320mm t=5mm
c-section channel	L=370mm Height=70mm Width=40mm
Total shell (duct)	L=370mm Height=80mm Width=320mm

Table 5. Dimensions of the wing box

All the above parts are joined using assembly in solid works. The shell after modelling in solidworks looks as shown in the below Fig.9.

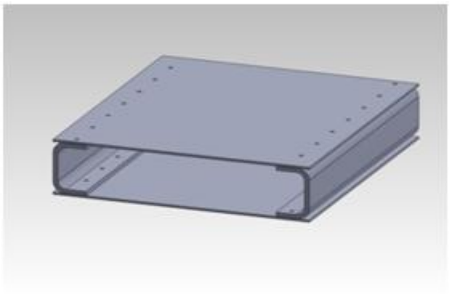

Fig.9Isometric view of model in Solidworks

1) Expand the modes and review results

The mode shapes corresponding to the frequencies can be visualized by selecting display option. The natural frequencies and the mode shapes are as follows. Here the first six modes are rigid modes i.e. corresponding to 3 translations and 3 rotations. Elastic modes are listed in the below table.

Mode	Frequency
7	65.745Hz
8	109.574 Hz
11	156.458 Hz
12	161.332 Hz
21	405.271 Hz

Table 6. Modal frequencies

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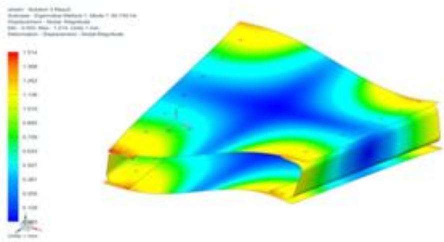

Fig.10 Mode shape-7

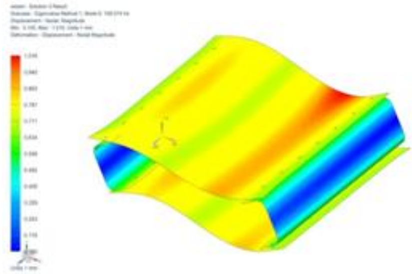

Fig.11 Mode shape-8

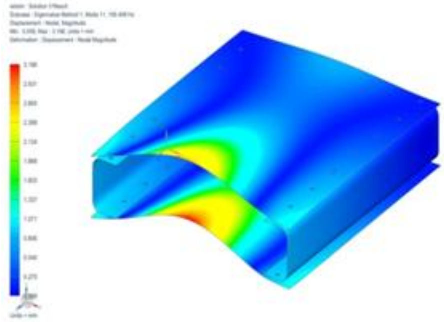

Fig.12 Mode shape-11

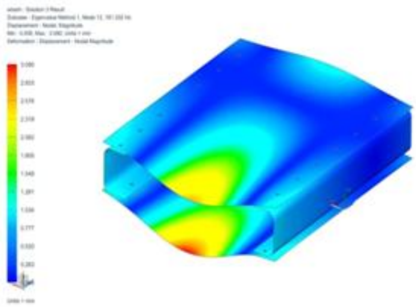

Fig.13Mode shape-12

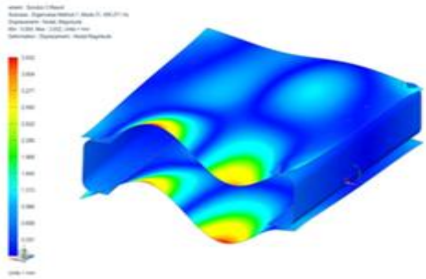

Fig.14 Mode shape-21

B. Experimental Verification

Experimental verification of the modal analysis of the shell structure is done by impact test method. The apparatus is discussed in next section.

1) Introduction to me-scope

The surface to be experimentally determined is designed in me-scope software

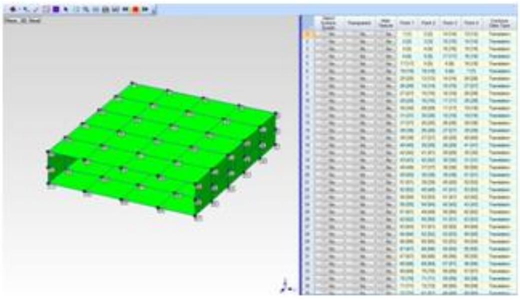

Fig.15 modelling in me-scope

2) Introduction to Experimental Apparatus

The experimental apparatus consists of:

1. Tri-axial sensor
2. Data acquisition unit
3. Impact test hammer
4. Bungee ropes and I-bolts
5. Cables

Fig.16 Impact hammerSet up

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Fig.17. Holding device

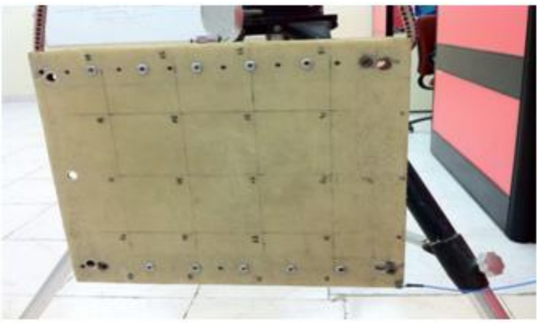


Fig.18 Sensor with axis shown

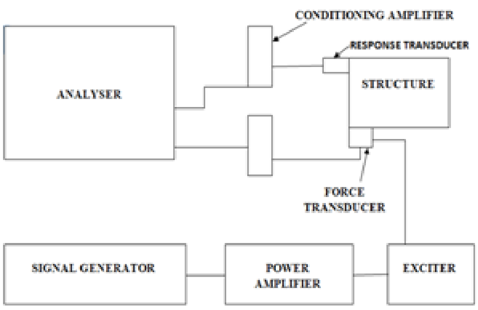


Fig.19 Working of setup

3) Impact test:

One of the most commonly used methods for measuring a system's natural frequency is to strike it with a mass and measure the response. This method is effective because the impact inputs a small amount of force in the requirement over a large frequency range. When performing this technique, it is important to consider impacting at different locations on the structure since all of a structure's resonant frequencies will always be measurable by impacting at any location and measuring at the same location. Both drive point and transfer point measurements should be taken when attempting to identify machine resonances. The entire shell structure is subdivided into 72 equidistant nodal points for performing impact hammer test. Data acquisition system consists of one tri-axial sensor is placed so that 3 responses are acquired for each of 38 nodal points (that is along 3 axis of the given nodal point on which we perform the impact test and acquire by means of three 3 data acquisition cables) and hence 72 X 3 = 216 FRF data were acquired.

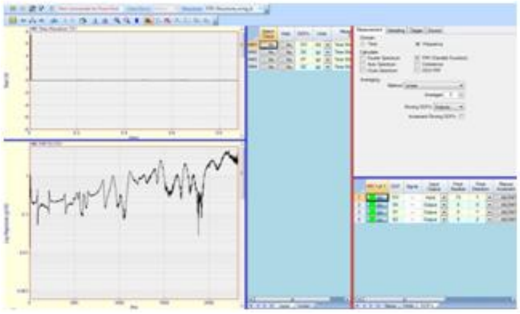


Fig.20 FRF curves as output

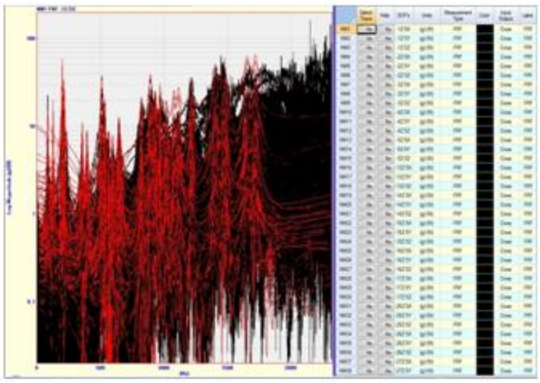


Fig.21 Graph

FFT of response vs frequency for tri-axial sensors (superimposition for all 216 frf)

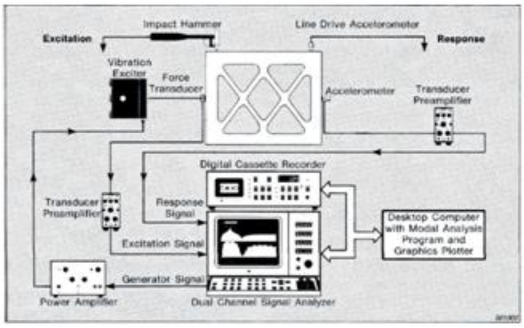


Fig.22 working of set up

one tri-axial sensor is placed so that one response is acquired. With the help of these responses the frequencies are identified as given below, which are obtained from the impact test. The various mode shapes obtained experimentally are shown in Fig.23,24,25& 26.

Frequency 1: 86.815Hz
Frequency 2: 129.62Hz
Frequency 3: 202.43Hz
Frequency 4: 217.21Hz
Frequency 5: 395.35Hz

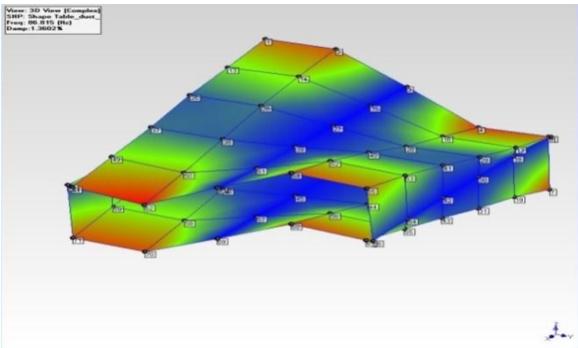


Fig.23 mode shape 7

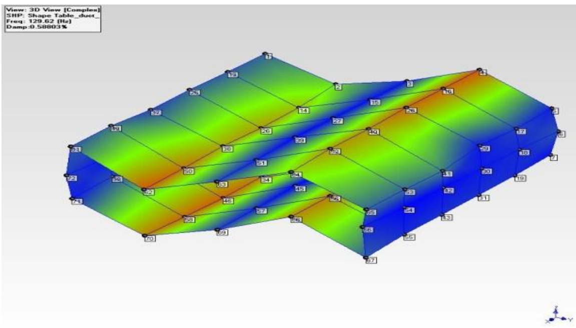


Fig.24 mode shape 8

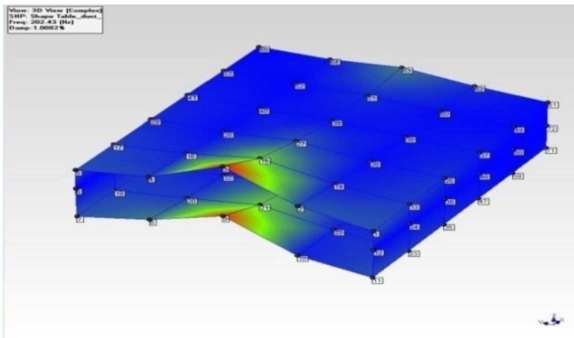


Fig.25 mode shape 11

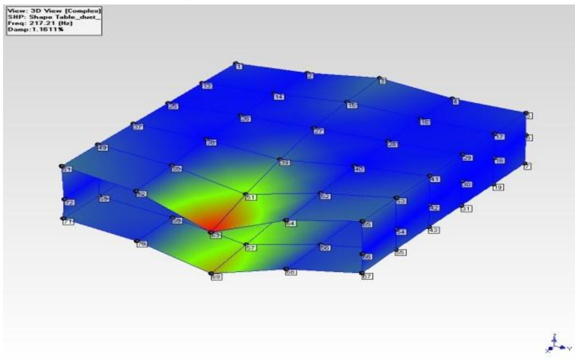
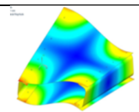
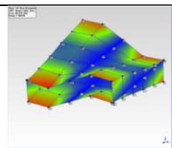
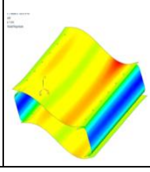
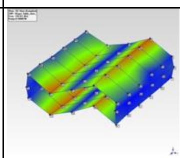


Fig.26 mode shape 21

C. RESULTS AND DISCUSSIONS

Comparison of computational and experimental analysis

Mode	computational Results	Mode shape	Exp Results	Mode shape
7	47.649 Hz		86.815 Hz	
8	109.574Hz		129.62 Hz	

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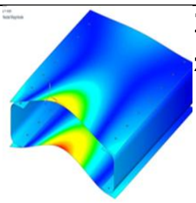
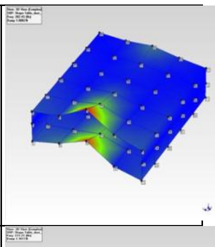
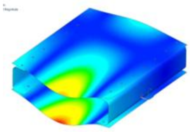
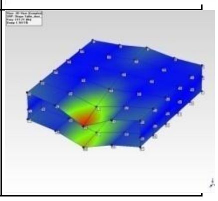
11	158.45 Hz		202.43Hz	
21	161.332 Hz		217.27 Hz	

Table 7. comparison of modes shapes

VII.CONCLUSION:

1. The frequency values are close enough as it is a composite structure and mode shapes are matched
2. The frequency of the composite wing box is higher in experimental compared to FEA this is because of behaviour of composite based on many external factors like layup and other factors.
3. During the experimental analysis the external forces have an effect.
4. However, the correlation of results and investigation is carried for further extension of project

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