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Aircraft Wing Weight Optimization by Composite Material Structure Design Configuration

R.Kirubakaran[1], D.Lokesharun[2], S.Rajkumar[3], R.Anand[4]
[1, 2, 3]*(Aeronautical Engineering, PARK College of Engineering and Technology / Anna University, India)*
[4]*(Aeronautical Engineering, Excel Engineering College / Anna University, India)*
Corresponding Author: R.Kirubakaran1.

Abstract : *High-performance composite materials exhibit both anisotropic strength and stiffness properties. These anisotropic properties can be used to produce highly-tailored aircraft structures that meet stringent performance requirements, but these properties also present unique challenges for analysis and design. Composite materials can provide a much better strength-to-weight ratio than metals. The lower weight results in lower fuel consumption and emissions and, because plastic structures need fewer riveted joints, enhanced aerodynamic efficiencies and lower manufacturing costs. This paper examines the design of metallic and composite aircraft wings in order to assess how the use of composites modifies the trade-off between structural weight and drag by comparing the numerical results of both metallic and composite designs by Studying the best possible combination of composite material for internal structure of wing, Design and analysis metallic and composite structures and finally finding the correct combination of material and ply orientation by optimizing the design to match the stress distribution. The above mentioned tasks is done with the help of CATIA V5 (Structural Design), Hyper mesh (Finite element modeling) and Ansys (solving). Optimum design is found by tabulating stress and displacement for each ply combination. Weight reduction of the wing is also found by comparing with the optimum composite wing with metallic wing.*

Keywords: *Composite, Optimization, Weight Reduction.*

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I. Introduction

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.

Composite structures typically consist of laminates stacked from layers with different fiber orientation angles. The layer thickness is normally fixed, and fibber orientation angles are often limited to a discrete set such as 0°, ±30°, ±45°, ±75°, and 90°. This leads to Different combinations of ply orientation and among that one will gives the better results, that is the optimized design For composite structures. A unidirectional laminate is a laminate, in which all fibers are oriented in the same direction, cross-ply laminate is a laminate in which the layers of unidirectional lamina are oriented at right angles to each other and quasi-isotropic laminate behaves similarly to an isotropic material; that is, the elastic properties are same in all direction. Unidirectional composite structures are acceptable only for carrying simple loads such as uni-axial tension or pure bending. In structures with complex requirements of loading and stiffness, composite structures including angle plies will be necessary. Since each laminate in the composite material can have distinct fibre orientations which may vary from the adjoining laminates, the optimum ply orientation is also obtained as a result of the parametric study conducted.

1.1 Wing

The wing plan form geometry for a transport aircraft is estimated using figure a.1 in appendix a. Figure a.1 shows that the wing does not have a straight taper, but a “kick” from the edge of the fuselage to the position where the engine attaches to the Wing. The wing plan form with the “kick” is illustrated in figure 1.1. The kick k, which is given as a fraction of the half span, is located where there is a geometry transition and the wing taper is discontinuous.

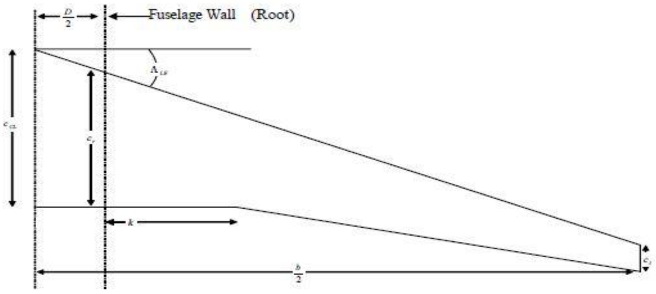


Fig. 1 Geometry for the wing for the wide body jet transport.

The known wing geometry includes the Aspect ratio AR, span b, and leading edge sweep Λ_{LE}. The wing reference surface area S_{ref} is determined from the aspect ratio and span. The span, reference surface area, and the chord distribution for the wide body are used to calculate the mean geometric chord c. The formula used to calculate mean geometric chord is shown. This chord distribution is determined from Figure A.1 in Appendix A. Since the chord distribution is known the root chord Cr and the tip chord c_t are known. The leading edge has a constant sweep. The quarter chord sweep $\Lambda_{c/4}$ is determined. The exposed wing reference area S_{exp} is equal to the reference wing area minus the referenced wing area within the fuselage diameter (or fuselage wall).

$$AR = \frac{b^2}{S_{ref}} \qquad \overline{c} = \frac{2}{S_{ref}} \int_0^{\frac{b}{2}} c^2 dy \qquad \tan \Lambda_{c/4} = \frac{3}{4} \tan \Lambda_{LE}$$

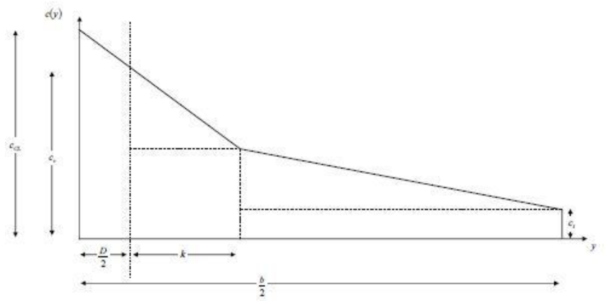


Fig. 2 Wing box structure

The wing box structure extends the span of the wing and the width of the wing box is a fraction of the chord for a specific wing station. The wing box is shown with in the geometry of the wing plan form. It is assumed for simplicity of analysis that the center of the wing box is located at the quarter chord of the wing plan form which is also assumed to be the location of the aerodynamic center and center of pressure. This implied that the aerodynamic resultant force is located at the center of the wing box and that there is no pitching moment on the wing. The configurations for the wide and narrow body transport structures are both two spar concepts with rib spacing indicated by the stiffened panel geometry length.

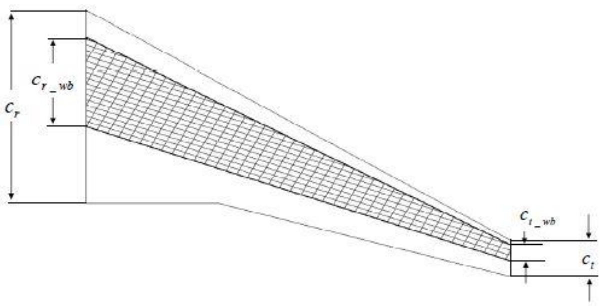


Fig. 3 Wing-box thickness and wing box with taper transition at the “kick”.

. This is an important detail to recognize because there is a large thickness taper from the root to the “kick” and a subtle thickness taper from the “kick” to the tip, which has a large effect on the running load profile distribution over the wing span. There is a beam that runs behind the rear landing gear, from the reference wing plan form centerline to the "kick" this beam is neglected in this analysis because the details needed to size this beam were not readily available.

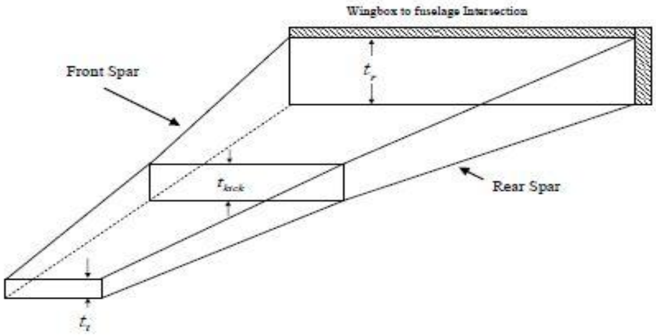


Fig. 4 Ribs and spars arrangements

1.2 Ribs

Fig. 5 Arrangements of ribs

In an aircraft, ribs are forming elements of the structure of a wing, especially in traditional construction. By analogy with the anatomical definition of "rib", the ribs attach to the main spar, and by being repeated at frequent intervals, form a skeletal shape for the wing. Usually ribs incorporate the airfoil shape of the wing, and the skin adopts this shape when stretched over the ribs. There are several types of ribs. Form-ribs, plate-type ribs, truss ribs, closed-ribs, forged ribs and milled ribs, where form-ribs are used for light to medium loading and milled ribs are as strong as it can get.

1.3 Spars

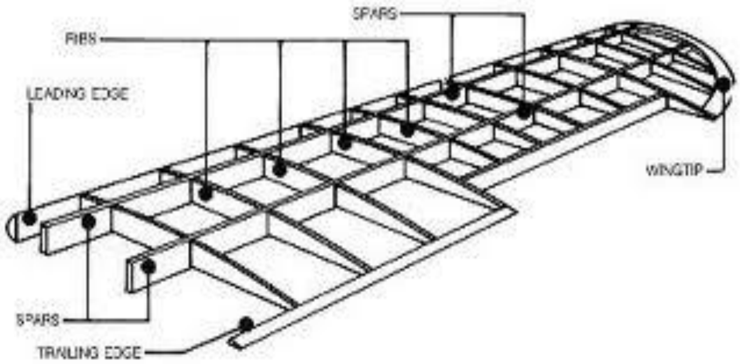


Fig. 6 spars arrangements

In a fixed-wing aircraft, the spar is often the main structural member of the wing, running span wise at right angles (or thereabouts depending on wing sweep) to the fuselage. The spar carries flight loads and the weight of the wings while on the ground. Other structural and forming members such as ribs may be attached to the spar or spars, with stressed skin construction also sharing the loads where it is used. There may be more than one spar in a wing or none at all. However, where a single spar carries the majority of the forces on it, it is known as the main spar.

1.4 Composites

Composite materials are becoming more important in the construction of aerospace structures. Aircraft parts made from composite materials, such as fairings, spoilers, and flight controls, were developed during the 1960s for their weight savings over aluminium parts. New generation large aircraft are designed with all composite fuselage and wing structures and the repair of these advanced composite materials requires an in-depth knowledge of composite structures, materials, and tooling. The primary advantages of composite materials are their high strength, relatively low weight, and corrosion.

1.5 Laminated Structures

Composite materials consist of a combination of materials that are mixed together to achieve specific structural properties. The individual materials do not dissolve or merge completely in the composite, but they act together as one. Normally, the components can be physically identified as they interface with one another. The properties of the composite material are superior to the properties of the individual materials from which it is constructed. An advanced composite material is made of a fibrous material embedded in a resin matrix, generally laminated with fibbers oriented in alternating directions to give the material strength and stiffness. Fibrous materials are not new; wood is the most common fibrous structural material known to man.

1.5.1 Applications of composites on aircraft include:

1. Fairings
2. Flight control surfaces
3. Landing gear doors
4. Leading and trailing edge panels on the wing and
5. stabilizer
6. Interior components
7. Floor beams and floor boards
8. Vertical and horizontal stabilizer primary structure on
9. large aircraft
10. Primary wing and fuselage structure on new generation
11. large aircraft
12. Turbine engine fan blades
13. Propellers

II. Design Of Composite Wing

2.1 Overview of Finite Element Analysis

Prior to the development of FEA the only way to validate a design or test a theory was to physically test a part. This was and still is both time consuming and expensive. While FEA will never replace the final physical testing and validation of a design, it can drastically reduce the time and money spent on intermediate stages and concepts. FEA in its infancy was limited to large scale computing platforms but the development of powerful personal computers combined with intuitive software packages. Such as Hyper Works have brought FEA to the engineers desktop and has broadened its use and accuracy many fold. Finite Element Analysis is now a vital and irreplaceable tool in many industries such as Automotive, Aerospace, Defense, Consumer Products, Medical, Oil and Gas, Architecture and many others. FEA is performed in three stages, Pre-Processing, Solving and Post Processing and those are outlined below.
Step 1: Pre-Processing, Step 2: Solving, Step 3: Post-Processing

2.2 Design of Composite Wing

In this project, The Composite wing model is designed using the CATIA V5 and Meshed by using the HYPERMESH.

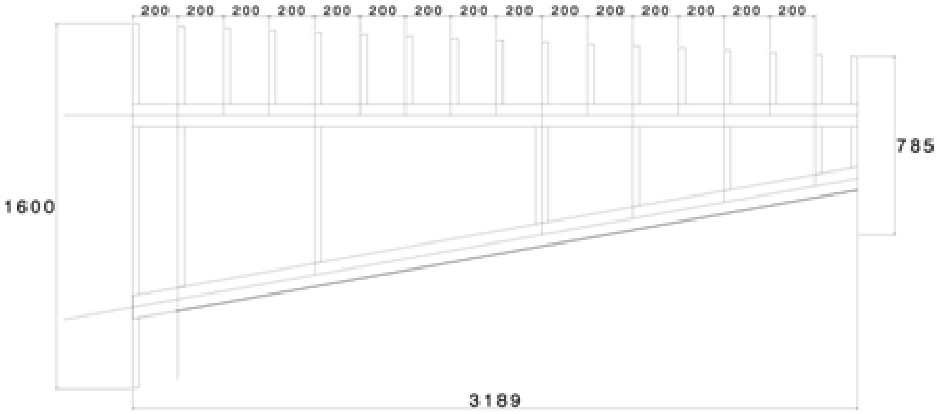
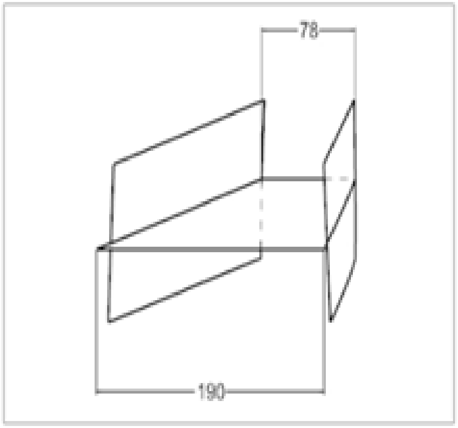


Fig. 7 Dimension of Wing



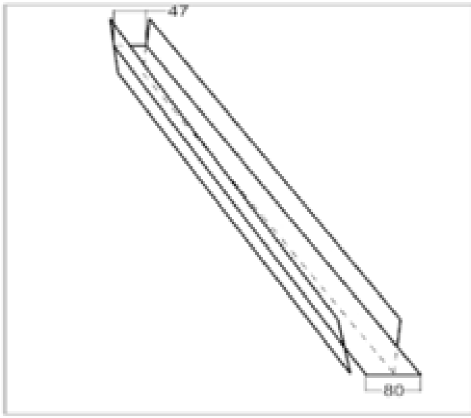


Fig. 8 Main Spar Dimension

Fig. 9 Rear Spar Dimension

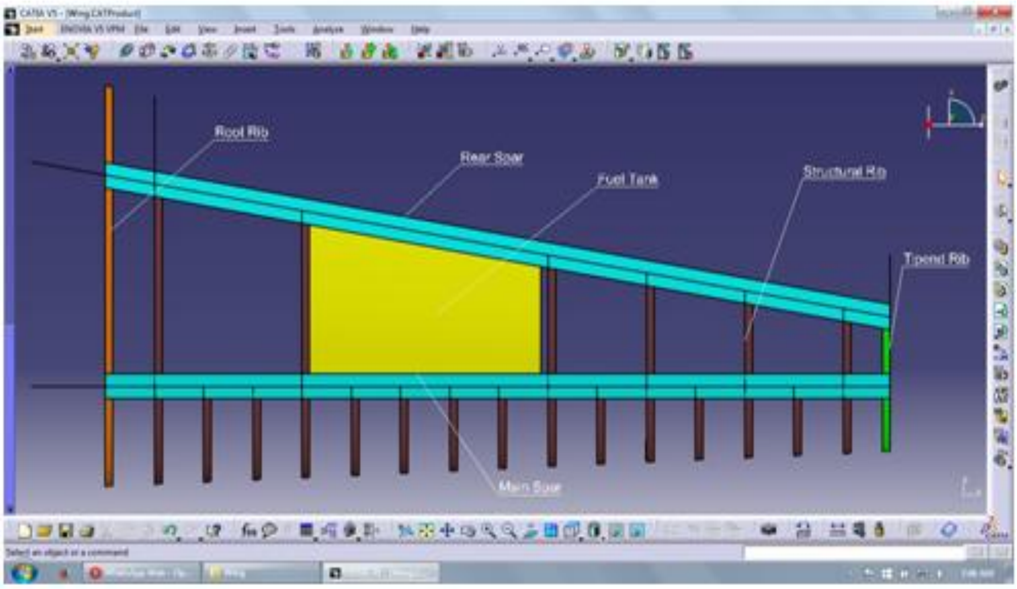


Fig. 10 Wing Detail

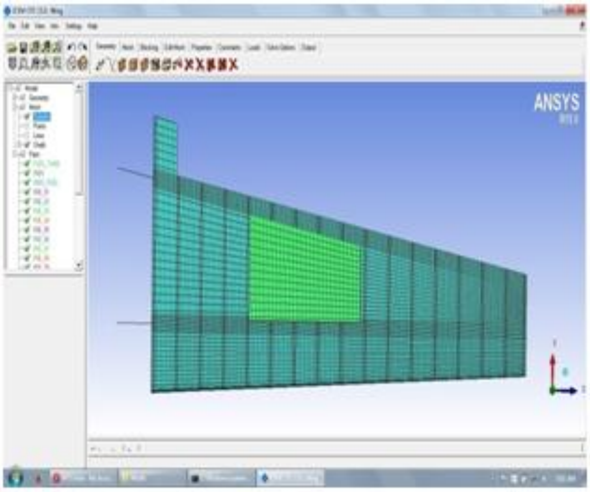


Fig. 11 Meshed Wing

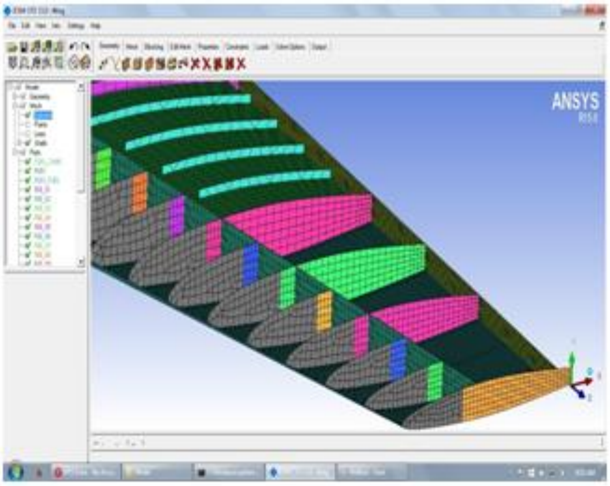


Fig. 12 Meshed Wing

III. Analysis Of Metallic And Composite Wings

3.1 Ply Details

Applied Pressure: 1280 Pa

Table 1 Ply Details of composite wing 1 and Composite wing 2

Ply Details of Composite Wing Analysis 1

Stack up 1
Material used:
Epoxy _EGlass_UD "Top-Down type"
Epoxy _EGlass_UD "Thickness: 0.8"
Epoxy _EGlass_UD "Weight/Area:1.6e-09"
Epoxy _EGlass_UD "No Symmetry"
ply Orientation: 0/-45/45/90

Stack up 2
Material used:
PVC_Foam
Epoxy _EGlass_UD "Top-Down type"
Epoxy _EGlass_UD Thickness: 19.8
Epoxy _EGlass_UD Weight/Area: 3.12e-09
Epoxy _EGlass_UD No Symmetry
Ply Orientation: 0/0/-45/45/90

Skin Bottom
Ply 1: Stack up 1
Ply 2: Stack up 1
Ply 3: Staly1: Stack up 1

Skin Top
Ply 1: Stack up 1
ply 2: Stack up 1

Ply Details of Composite Wing Analysis 2

Stack up 1
Material used:
Epoxy_Carbon_UD "Top-Down type"
Epoxy _EGlass_UD "Thickness: 0.6"
Epoxy _EGlass_UD "Weight / Area: 1.10e-09 Epoxy"
Epoxy _EGlass_UD "No Symmetry"
Ply Orientation : 0/-45/45

Stack up 2
Material used:
Honey comb core "Top-Down type"
Epoxy _EGlass_UD "Thickness: 3.6"
Epoxy _EGlass_UD "Weight/Area: 1.30e-09"
Epoxy_Carbon_UD "No Symmetry"
Ply Orientation: 0/0/-45/45/90

Stack up 3
Material used:
PVC_Foam
Epoxy _EGlass_UD "Top-Down type"
Epoxy _EGlass_UD "Thickness: 19.8"
Epoxy _EGlass_UD "Weight/Area: 3.02e-09"
Epoxy_Carbon_UD "No Symmetry"
Ply Orientation: 0/0/-45/45/90

Skin Bottom
Ply 1: Stack up 1
Ply 2: Honey Comb with 0 Deg
Ply 3: Stack up 1

Skin Top
Ply 1: Stack up 1
ply 2: Honey Comb with 0 Deg
ply 3: Stack up 1

Main Spar Bottom Cop
Ply 1: Stack up 1

Rear Spar Cap
Ply 1: Stack up 1

Main & Rear Spar Top
Ply 1: Stack up 1

Main Spar 1 & 2
Ply 1: Stack up 1

Rear Spar 1 & 2
Ply 1: Stack up 1

Ribs
Ply 1: Stack up 1
Ply 2: Stack up 1

Fuel Ribs & Fuel Tank
Ply 1: Stack up 1

Main Spar Bottom Cop
Ply 1: Stack up 1

Rear Spar Cap
Ply 1: Stack up 1
Main & Rear Spar Top
Ply 1: Stack up 1

Main Spar 1 & 2
Ply 1: Stack up 2

Rear Spar 1 & 2
Ply 1: Stack up 2

Ribs
Ply 1: Stack up 1
Ply 2: Honey Comb with 0 Deg
Ply 3: Stack up 1

Fuel Ribs & Fuel Tank
Ply 1: Stack up 1
Ply 2: Honey Comb with 0 Deg
Ply3: Stack up 1

3.2 Material Properties for Composite wing Analysis

Table 2 Material Properties

Material / Properties	Density kg/m^3	E_1 N/m^2	E_2 N/m^2	E_3 N/m^2	ν_{12}	ν_{13}	ν_{23}	G_{12} N/m^2	G_{31} N/m^2	G_{23} N/m^2
Epoxy-EGlass UD	2e-09	45000	10000	10000	0.3	0.3	0.3	5000	5000	3846.1
PVC Foam	8e-11	103	103	103	0.3	0.3	0.3	39.2	39.2	39.2
Honey Comb	8e-11	1	1	1	0.3	0.3	0.3	1e-06	70	37

3.3 Metallic Wing Properties

Material Used: Aluminum Alloy

Tensile Yield Strength: 280MPa

Tensile Ultimate Strength: 310MPa

Density: 2770 kg/m^3

IV. Result And Discussion

4.1 Result of Composite Wing 1

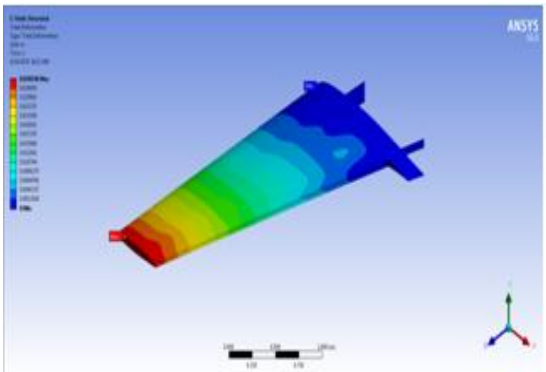


Fig. 13 Total Deformation

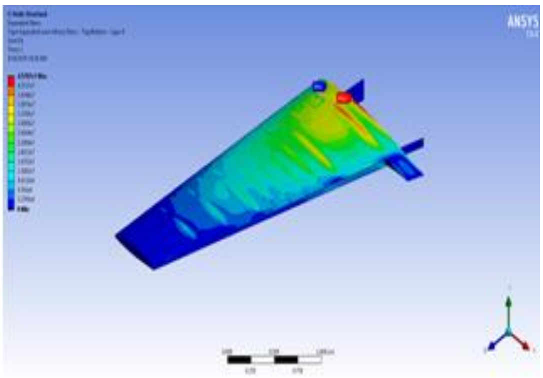


Fig. 14 Equivalent Stress

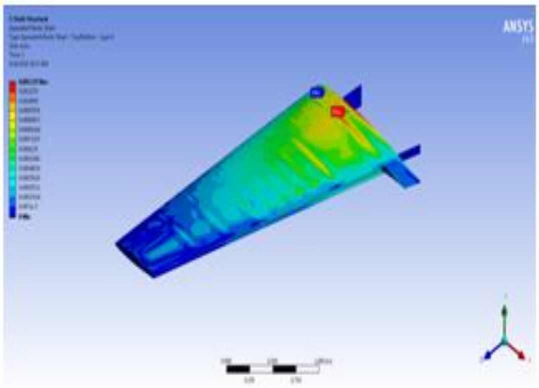


Fig. 15 Equivalent Elastic Strain

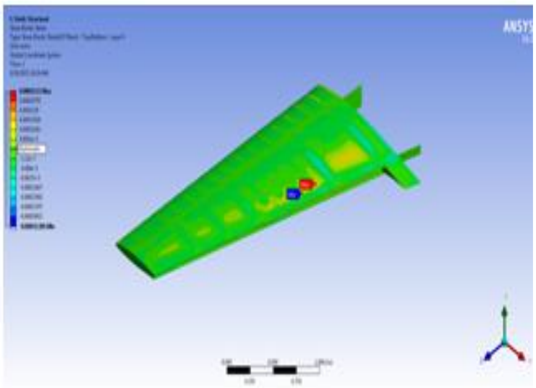


Fig. 16 Shear Elastic Strain

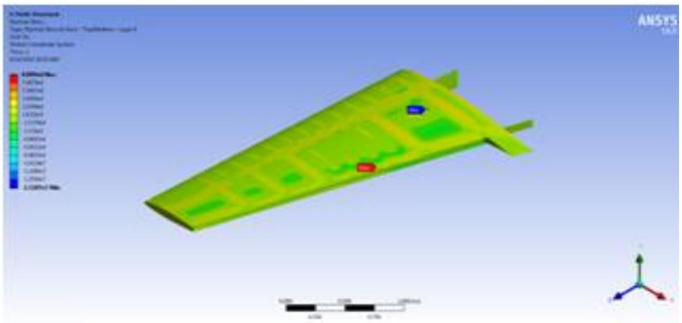


Fig. 17 Normal Stress

4.2 Result of Composite Wing 2

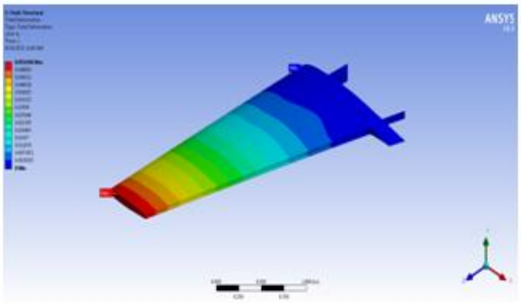


Fig. 19 Total Deformation

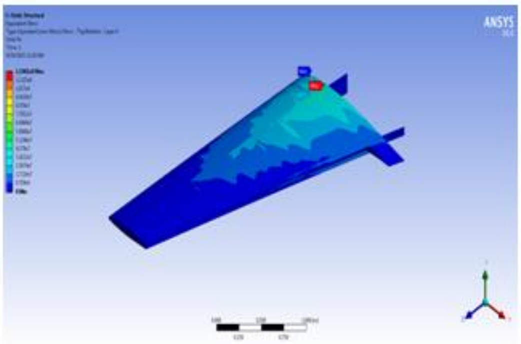


Fig. 18 Total Deformation

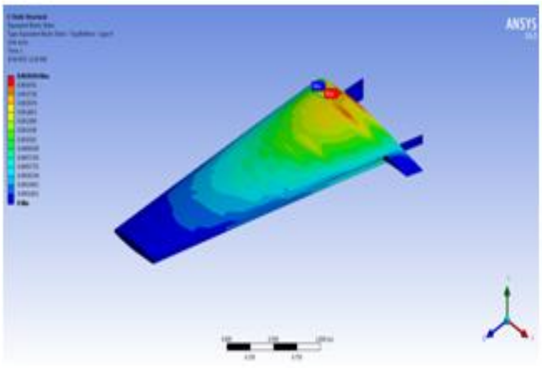


Fig. 20 Equivalent Elastic Strain

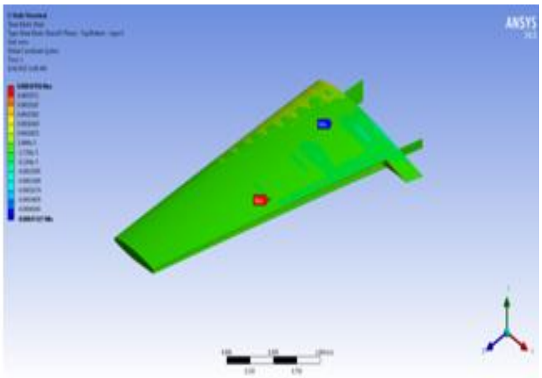


Fig. 21 Shear Elastic Strain

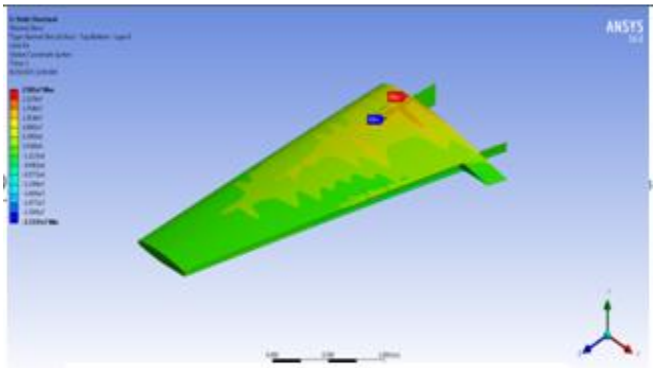


Fig. 22 Normal Stress

4.3 Result of Metallic Wing

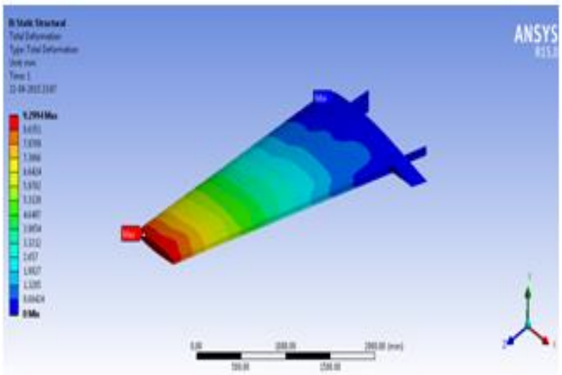


Fig. 23 Total Deformation

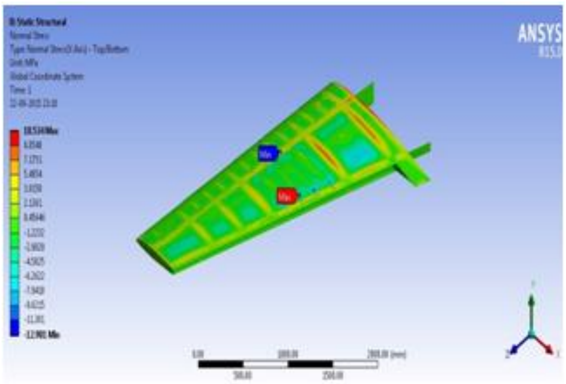


Fig. 24 Equivalent Stress

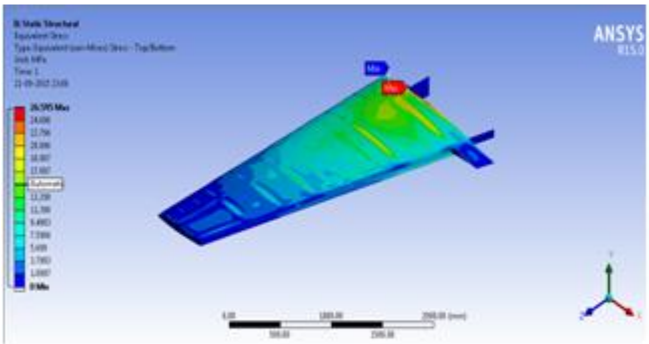


Fig. 25 Equivalent Elastic Strain

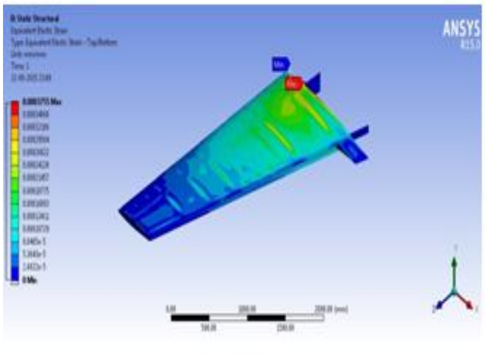


Fig. 26 Shear Elastic Strain

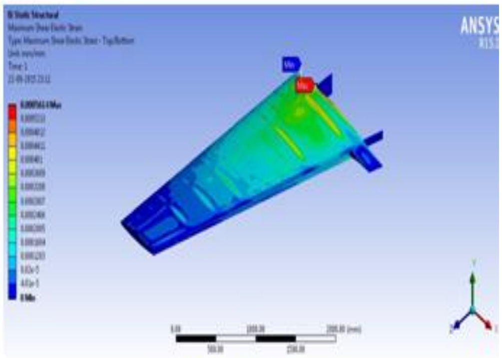


Fig. 27 Normal Stress

V. Conclusion

Table 3 Comparison of Structural Values

S. No	Variants	Metallic Wing (Max. Value)	Composite Wing 1 (Max. Value)	Composite wing 2 (Max. Value)
1	Total Deformation	9.2994 mm	30.196 mm	51.696 mm
2	Equivalent Stress	26.595 MPa	45.787 MPa	119.81 MPa
3	Equivalent Elastic Strain	0.00037	0.001247	0.002020
4	Normal Stress	10.534 MPa	8.809 MPa	25.01 MPa
5	Shear Elastic Strain	0.000561	0.000312	0.000419
6	Weight	57.3 kg	36.4 kg	27.9 kg

From the Comparison of above noted data, The Composite Wing 2 of three stackups with different ply orientation (Stackup 1: 0/-45/45, Stackup 2: 0/0/-45/45/90, Stackup 3: 0/0/-45/45/90) give the best mechanical properties than others.

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