

Design and Analysis of Wing of an Ultralight Aircraft

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ABSTRACT: The paper deals with the structural design and analysis of high wing of an ultralight aircraft. The wing design involves its initial considerations like planform selection, location to the aircraft and the structural design involves the design calculations for the selection of airfoil, area of the wing, wing loading characteristics and weight of the wing. The design is done corresponding to the calculated values with the help of designing software CATIA and the analysis is done to show the structural deformations and stress for the applied loading conditions with the help of ANSYS 14.0, also the drag polar for the applied flow conditions is shown with the help of ANSYS FLUENT a flow analysis software. The objective of this project is to compare the results obtained for different materials like Al 2024-T3, Al 6061-T6, Al 7075-T651 & Al 7075 + 15% FLY ASH MMC using analysis software. From the results we will conclude which material is having better properties.

KEYWORDS: Ultralight Aircraft, Wing Structural Design, Structural and flow analysis, CATIA, ANSYS..

I. INTRODUCTION

i. Wing and its structure

Wings develop the major portion of the lift of a heavier-than-air aircraft. Wing structures carry some of the heavier loads found in the aircraft structure. The particular design of a wing depends on many factors, such as the size, weight, speed, rate of climb, and use of the aircraft. The wing must be constructed so that it holds its aerodynamics shape under the extreme stresses of combat manoeuvres or wing loading. Wing construction is similar in most modern aircraft. In its simplest form, the wing is a framework made up of spars and ribs and covered with metal.

Spars are the main structural members of the wing. They extend from the fuselage to the tip of the wing. All the load carried by the wing is taken up by the spars. The spars are designed to have great bending strength. Ribs give the wing section its shape, and they transmit the air load from the wing covering to the spars. Ribs extend from the leading edge to the trailing edge of the wing. In addition to the main spars, some wings have a false spar to support the ailerons and flaps. Most aircraft wings have a removable tip, which streamlines the outer end of the wing.

ii. Ultralight Aircraft:

Ultralight aircraft are generally called microlight aircraft whose definitions, the weight and the speed limits differ from country to country. Ultralight aircraft, as a group, are designed primarily for recreational flying for distances of not more than 165.4 kilometers from a home base. However, recent several models have been developed to include aerobatic flying and have been considered by the military for front line reconnaissance.

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Fig. 1. An ultralight aircraft

Ultralight aircraft designs, which are classified by the type of structure:

- The first generation ultralights were actually “hang gliders” with small engines added to them, for self-launching.
- The second generation ultralights are powered aircraft having "2-Axis" control systems.

The third generation ultralights have strut-braced wings and airframe structure use “3-Axis” control systems.

II. RELATED WORK

T.V. Baughn and P.F. Packman [1] in 1986, conducted a finite element analysis to determine the structural integrity of a high-wing cable-supported ultralight aircraft. A simple, symmetrical, half-structure macro-model was analyzed and subjected to level flight loading and two-wheel-landing loading conditions. Flexural and bending stiffness for the supported and unsupported wing were also determined. A preliminary damage tolerance analysis was conducted in which selected cable elements and wing compression struts were removed, the redistributed loads calculated, and possible aircraft flight configurations examined. The model can generate all cable loads, displacement of each structural node (for each loading condition), generate displacement plots, and locate potential highly stressed regions.

Baughn, T. and Johnson, D. [2] in the same year of 1986, proposed a design change from high-wing cable-supported to strut supported aircraft. One of the most common designs is the high wing cable supported ultralight. Because of its simple shape and method of construction owners like to modify the structure and aerodynamic surfaces to attempt to improve the performance of the aircraft. One of the more common modification requests is for the conversion from a cable supported to a strut supported aircraft. The objective of the modification is to reduce the drag and improve the performance of the ultralight. The purpose of their study is to determine the structural performance of the cable supported aircraft and compare it to the structural performance of a strut supported version of the same aircraft and to provide an estimate of the change in drag associated with the conversion from cable supported to strut supported.

Girish S. Kulkarni [3] in 1987, with the help of all the design guidelines provided by Baughn, T along with considering critical condition in unaccelerated flight, done a Finite element method based structural design to analyze the behaviour of an airplane under Aerodynamic loading.

Zdobyslaw Goraj [4] in 2000, done a conceptual design of main wing, body and empennage for high altitude long endurance aerial vehicle. An arrangement of main spar, ribs, shell and strut for high aspect ratio main wing and tailplane has been proposed. A number of characteristics (stiffnesses, mass distributions, moments of inertia etc), necessary for flutter calculations, is included in this paper. The structural characteristics are computed using

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NASTRAN programme. The critical flutter speed for empennage has been found by means of the conventional U-G method. The Doublet-Point-Method (DPM) for non-coplanar configuration was used to compute the unsteady aerodynamic forces. Aerodynamic model of empennage includes 168 aerodynamic panels. This analysis can be treated as a starting point for further wing optimisation. The main goal is to obtain the structure lighter and aerodynamically more efficient - the feature - being very important in long endurance missions.

William Zimmerman and Howard W. Smith [5] in 2001, investigated about the growing need for more stringent design criteria for ultralight aircraft. Unlike most general aviation aircraft, the ultralight lacks sufficient design criteria and more importantly it lacks sufficient certification enforcement.

L. Pascale & F. Nicolosi [6] in 2006, proposed a new design which is based on the idea to build a 4-seat aircraft with two light engines (Rotax 912S, usually used for ultralight aircraft) and to enter the market with a twin-engine aircraft with the same weight of a single engine aircraft. The present paper shows all main criteria on which the design of the aircraft and the choice of the configuration have been based. At Dipartimento di Ingegneria Aerospaziale (DIAS) of University of Napoli "Federico II" a deep aircraft aerodynamic investigation has been performed both numerically and experimentally through an extensive wind-tunnel test campaign. All tests and research activities have been focused on the analysis and optimization of aircraft aerodynamics. Detailed measurements of fuselage and nacelle aerodynamic effects. Design and tests of winglets to improve rate of climb in OEI (One Engine Inoperative) condition.

Huiwen Hu and Huaian Kao [7] in 2009, presented the validation of finite element model (FEM) of an ultralight aircraft structure by using experimental modal analysis (EMA). The main structure of ultralight aircraft consists of the wings, the fuselage and the empennage structures, which are fabricated by means of aluminum tubes through the bolts, rivets and the brackets. A commercial code, ANSYS, is used to establish the FEM. Normal mode analysis is performed to obtain the natural frequencies and mode shapes under a completely free boundary condition. EMA is conducted on the wings, the fuselage and the integral structures to obtain their natural frequencies and mode shapes, respectively. The FEM is then validated and updated according to the correlation of natural frequencies and mode shapes between EMA and FEA. It is essential to assure that the FEM is equivalent to the real aircraft structure for the further structural analysis.

Kesavulu A, F. Anand Raju, & Dr. M.L.S. Deva Kumar [8] in 2014 published a paper on Metal matrix composites [MMC]. MMC are most important materials used for recent works in the industry and engineering applications. Fly ash particles are used in metal matrix composites, are low cost and low density are available in large quantities of waste by product in power plants. The adding of fly ash with aluminium reinforcement by using stir casting process it can reduce the cost and density of aluminium material. Metal composite processes are improved mechanical properties like strength, hardness, low density and good wear resistance compared to other metals. In their study, aluminium clad and fly ash chemical analysis is studied before and after mixing and forming as particulate metal matrix composite and comparing the mechanical, physical properties of the MMC at varying % of fly ash addition. By comparison at various levels 15% of fly ash the MMC gives good mechanical and physical properties. This type of MMC is widely useful in light weight vehicles and aerospace application.

The ultralight aircraft differs from country to country based on speed and weight. During 19th century the ultralight aircraft weight ranges above 1000 kg. Now a days, the weight of ultralight varies from 155kg in USA to 750 kg in Brazil. But in India, the gross weight of ultralight aircraft is 450 kg.

III. WING DESIGN

i. Wing Design Parameters

The design parameter like wing span, wing twist, taper ratio, wing sweep, wing thickness, wing aspect ratio and wing dihedral are calculated and the wing is designed in CATIA R20. Actual wing area can be calculated from the total takeoff weight and the actual wing loading values. After comparing the wing loading for different flight conditions such as stall, cruise it is found that the minimum wing loading is obtained from the stall constraint. This value is taken to be the actual wing loading value.

$$\text{Total takeoff weight, } W_0 = 300.61\text{kg}$$

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The actual wing loading, $W/S = 4.4187 \text{ lbs/ft}^2 = 1.6 \text{ kg/m}^2$

$$\begin{aligned} \text{Wing Area} \quad S &= W_0 / (W/S) & (1) \\ &= 662.75 / 4.4187 \\ &= 149.98 \text{ ft}^2 \approx 150 \text{ ft}^2 \\ &S = 13.935 \text{ m}^2 \end{aligned}$$

This is the actual wing area for our design process
Then the wing span can be calculated as follows

$$\begin{aligned} b &= (S \cdot A.R.)^{1/2} & (2) \\ b &= (150 \cdot 6)^{1/2} \\ b &= 9.144 \text{ m} \end{aligned}$$

The half span value is, $b/2 = 4.572 \text{ m}$

Chord length can be calculated from,

$$\begin{aligned} c &= S/b & (3) \\ c &= (150/30) \\ c &= 1.524 \text{ m} \end{aligned}$$

The mean aerodynamic chord is calculated by

$$\begin{aligned} \text{M.A.C} &= 2/S \int_0^{b/2} c^2 dy & (4) \\ &= (2/150) \cdot (25 \cdot 15) \\ \text{M.A.C} &= 4.99875 \text{ ft} = 1.524 \text{ m} \end{aligned}$$

Weight of the wing

$$(W)_{\text{Wing}} = 96.948 \{ [WT0 \cdot n / 105] [A.R. / \cos(1/4)] \cdot 0.57 [S_w / 100] [(1+\lambda)/2(t/c)] \cdot 0.36 + [1 + V_{\text{cruise}}/500] \cdot 0.5 \} \cdot 0.993 \quad (5)$$

V_{cruise} = cruise airspeed at S.L. in knots = 45.59 m/s (or) 88.433 knots

$WT0$ = take-off weight in lbs = 662.948 lb = 300.68

n = ultimate load factor

$1/4$ = wing quarter chord sweep

S_w = wing area in $\text{ft}^2 = 150$

λ = taper ratio = 1

t/c = maximum thickness ratio = 11.725

$A.R.$ = 6

We have the values to substitute in the formula except load factor, it can be find out from the following procedure,

$$\begin{aligned} n &= L/W = (q_{\infty} \cdot S \cdot C_{L_{\text{max}}} / W) & (6) \\ n &= (q_{\infty} \cdot C_{L_{\text{max}}} / (W/S)) \\ n &= (1/2 \rho_{\infty} (V_{\text{stall}})^2 C_{L_{\text{max}}} / 4.4187) \\ n &= (3.2739(1.35)) / (4.4187) \\ n &= 1.00024 \approx 1 \end{aligned}$$

The load factor value has been obtained as 1 now we are going to substitute all the above values to find out the weight of wing.

$$(W)_{\text{Wing}} = 110.84 \text{ lbs}$$

$$(W)_{\text{Wing}} = 50.27 \text{ kg}$$

The above expression is valid for light conventional metal airplanes. Ultralight airplanes use 'Dacron' as the skin material, fly considerably lower-speeds. Therefore, it was decided to reduce the weight by say 23%

Now the weight of the wing will be reduced by 25.49 lbs

$$(W)_{\text{Wing}} = 110.84 - 25.49$$

$$(W)_{\text{Wing}} = 85.35 \text{ lbs}$$

$$(W)_{\text{Wing}} = 38.714 \text{ kg}$$

The below parameters are needed to design in CATIA

Wing span	= 9.144 m
Wing chord	= 1.524 m
Rib thickness	= 4.572 mm
Number of ribs	= 15
Spar thickness	= 3.0 mm
Number of spars	= 2
Skin thickness	= 1mm
The distance between each rib	= $9144/15 = 609.6 \text{ mm}$

Thickness of all rib is same which is equal to 4.572 mm, and this value is taken from the survey of rib design for ultralight aircraft's wing having almost same specifications. Clark Y is the name of a particular aerofoil profile, widely used in general purpose aircraft designs, and much studied in aerodynamics over the years. The airfoil has a thickness of 11.7 percent and is flat on the lower surface from 30 percent of chord back. The flat bottom simplifies angle measurements on propellers, and makes for easy construction of wings on a flat surface. Clark-Y airfoil coordinates are used to design the wing.

ii. Wing design in CATIA

Fig. 2. Images from CATIA (a) Sketch of the airfoil (b) Rib sections with spar (c) Wing section with skin

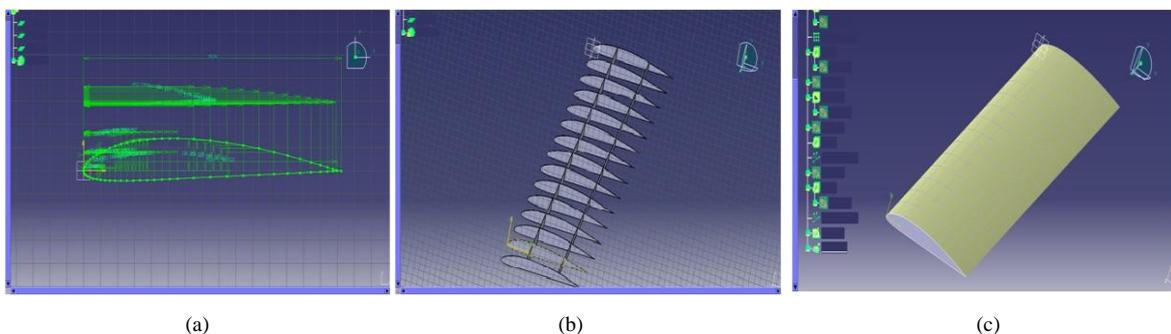


Fig. (a) shows the vertices and edges of the CLARK Y airfoil, (b) explains how we have done the design of rib and spar for the required dimensions & (c) shows the wing covered with skin.

IV. WING ANALYSIS

i. Mesh

In our analysis we have used Hybrid mesh. A hybrid mesh is a mesh that contains structured portions and unstructured portions. Normally unstructured mesh requires very less input, so everyone can use it. And structured mesh is used to mesh complex structure such as wing ribs, turbine and compressor blades. Hybrid mesh makes use of both structured and unstructured mesh, it automatically decide structured mesh in normal area and unstructured mesh in some complex area.

Fig. 3. Mesh model (a) for Flow analysis (b) for Structural Analysis with Skin (c) for Structural analysis without skin

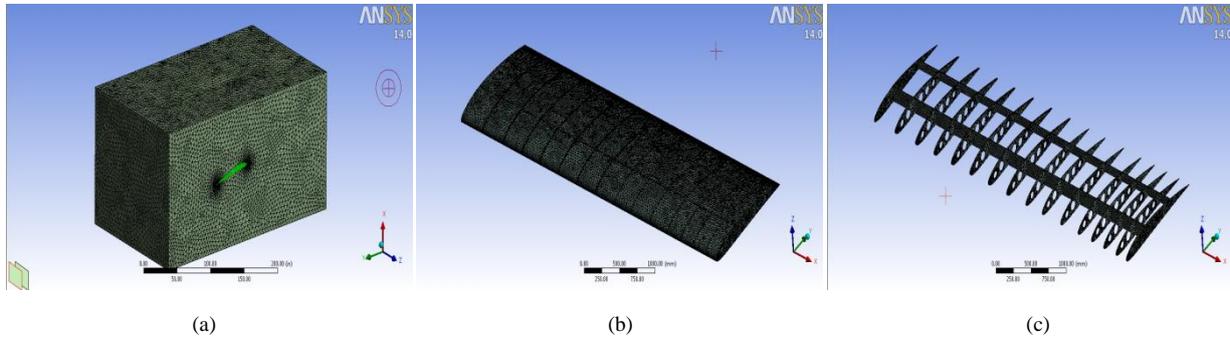


Fig. (a) shows mesh with No. of nodes= 221821 & No. of elements= 1251417, (b) mesh with No. of nodes= 124944 & No. of elements= 86897 and (c) with No. of nodes= 94297 & No. of elements= 42964.

ii. Computational Fluid Dynamics(CFD) Analysis

Boundaries for CFD analysis are considered as wall, inlet and outlet. The inlet is considered as velocity inlet and outlet is considered as pressure outlet and all other faces are considered as wall.

Air is taken as fluid for flow analysis. Density of the air is taken as 1.225kg/m³. The air velocity over the section is taken from the cruise condition. Horizontal axis shows the number of iterations. The meshed wing design is given specified boundary condition, that wall face and inlet section of 3D model and out let of exit section. The wing model applies some boundary condition in gambit. The velocity input in the inlet is given as 105 m/s max.

Lift Coefficient,

$$W = 1/2 * \rho_{\infty} * (v_{stall})^2 * S * C_{L_{max}} \tag{7}$$

Here the (W) is the weight of the aircraft at climbing condition the takeoff weight has been consumed then,

$$W = 300.63(0.97)$$

$$W = 291.61\text{kg}$$

V stall = 18m/s from the mission requirements

$$\rho_{\infty} = 1.225 \text{ kg/m}^3$$

S = 13 m² it is assumed for initial calculations from survey of microlight aircrafts under Indian Civil aviation requirements.

The required C_{L_{max}} value

Now by substituting the above values in the equation we can find the required C_{L_{max}} value

$$\begin{aligned} C_{L_{max}} &= 2859.821 / (1/2) * (1.225) * (18)^2 * (13) \\ &= 2859.821 * 2 / (1.225) * (18)^2 * (13) \\ &= 5719.642 / (5159.7) \end{aligned}$$

$$C_{L_{max}} = 1.1085$$

The C_{L_{max}} which has been obtained is only for three dimensional wing sections, we want the required Clmax for two dimensional airfoil profiles according to Raymers suggestions we can convert the 3D C_{L_{max}} to 2D C_{L_{max}}

$$\begin{aligned} C_{L_{max}} &= 0.9 C_{l_{max}} \\ 1.1085 &= 0.9 C_{l_{max}} \end{aligned} \tag{8}$$

$$C_{lmax} = 1.1085/0.9$$

$$C_{lmax} = 1.231$$

Drag Coefficient,

$$C_D = C_{D0} + K C_L^2 \tag{9}$$

Where, C_{D0} = zero lift drag coefficient

$K C_L^2$ = induced drag coefficient

$$C_{D0} = (S_{wet}/S) * C_{fe} \tag{10}$$

S_{wet}/S is the wetted surface area divided by the actual surface area of the wing section it is taken from the survey of ultralight aircrafts; it is ranging from 3 to 4 we have taken as 3.5 under assumption.

$$C_{fe} = 1.328/\sqrt{8} * 10^5 \tag{11}$$

Where the Reynolds number is taken as 800000 because at this Reynolds number we have got the maximum lift coefficient and stall angle as what was used earlier.

$$C_{fe} = 0.001484$$

$$C_{D0} = 3.5(0.001484)$$

$$C_{D0} = 0.00519$$

$$K C_L = (C_L^2) / (\pi e AR) \tag{12}$$

$$K C_L^2 = (1.35)^2 / (\pi * (0.95) * (6))$$

$$K C_L^2 = (1.8225) / (17.907)$$

$$K C_L^2 = 0.1017$$

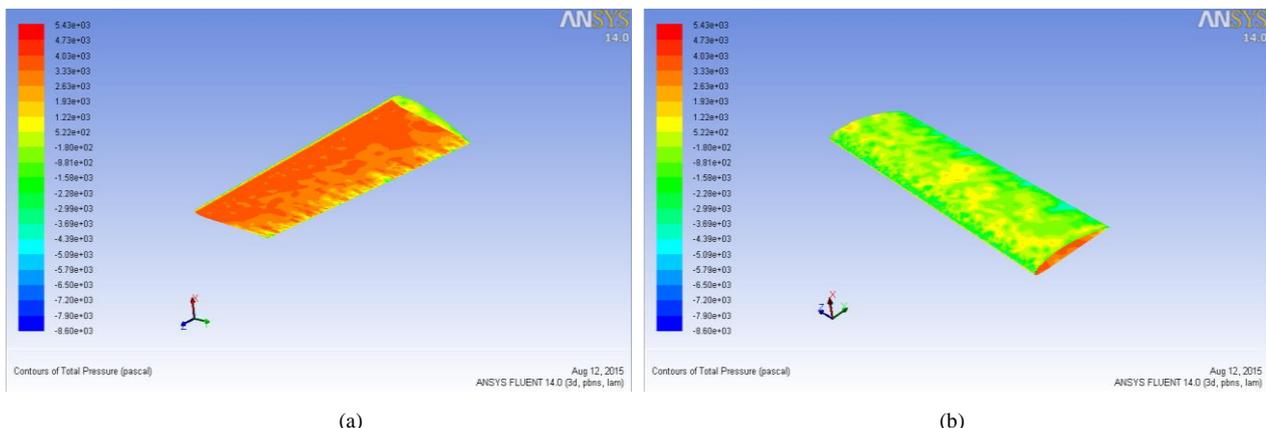
Now by substituting these values we can have the drag coefficient as

$$C_D = 0.00519 + 0.1017$$

$$C_D = 0.10696$$

$$C_D \approx 0.107$$

Fig. 4. Total pressure (a) on the bottom portion of wing (b) on the top portion of the wing



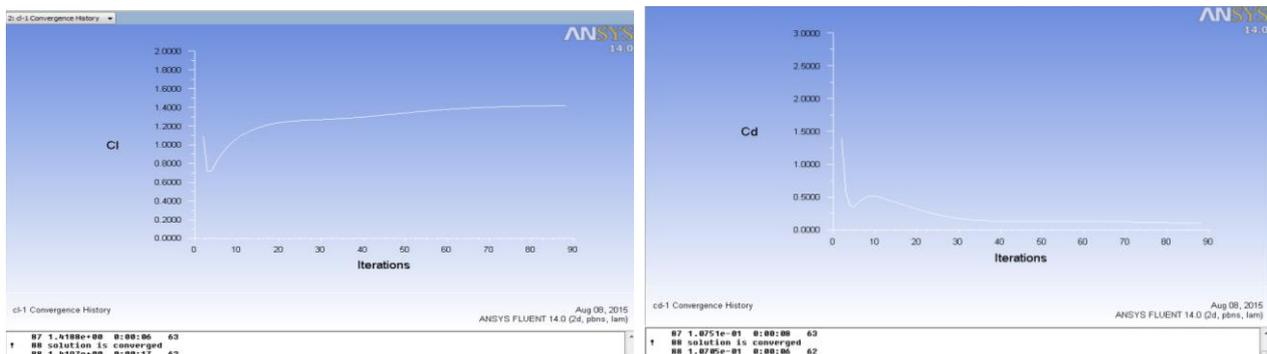
(a)

(b)

Fig (a) shows the maximum Pressure acting on the bottom portion of the wing is 5.43×10^3 Pascal & (b) shows the maximum Pressure acting on the top portion of the wing is 1×10^2 Pascal.

As the pressure on the bottom region is greater than the pressure on the top region of the wing, Lift is produced. Cross section of the wing is taken and C_L & C_D graphs are plotted as shown in Fig. 5.

Fig. 5. Convergence (a) of Lift Coefficient (b) of Drag Coefficient.



(a)

(b)

Fig. (a) shows coefficient of Lift value for the wing obtained through analysis is 1.4197 & Fig. (b) shows coefficient of drag value for the wing obtained through analysis is 0.107.

The results were obtained from the CFD for the wing in form of graph. From the graphs we came to know that lift achieved is as per our theoretical calculations. So our design and the CFD analysis is correct. The CFD result of the wing and graph CL and CD are plotted. From this analysis we can get the pressure load for the structural analysis of the wing, we can apply this result to structural analysis using software ANSYS FLUENT 14.0

iii. Structural Analysis

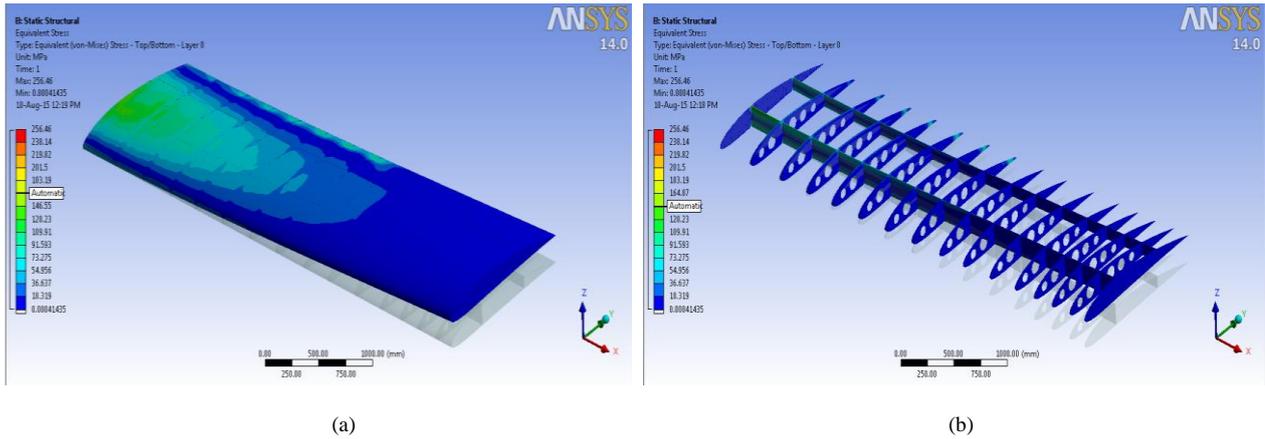
In this structural analysis, the chord area is fixed as a cantilever beam and pressure load is applied. The pressure force is imported from the results from the CFD analysis. The pressure that flow over the wing in the CFD analysis is applied as the pressure load on the top and bottom surface of the wing. This is done using software ANSYS FLUENT 14.0. The analysis is done for Al 2024-T3, Al 6061-T6, Al 7075-T651 & Al 7075+15% Fly Ash MMC having the properties as follows.

Table 1 Material properties

Material	Young's Modulus E (GPa)	Poisson's Ratio - μ	Ultimate Tensile Strength (Mpa)	Tensile Yield Strength (Mpa)	Density (Kg/m3)
Al 2024-T3	73.1	0.33	483	345	2 767.99
Al 6061-T6	68.9	0.33	310	276	2 698.79
Al 7075-T651	71.7	0.33	572	503	2 823.35
Al 7075+15%Fly Ash	71	0.3	591.3	469.7	2809.51

The structural analysis of ultralight aircraft wing has variation in deformation, strain and Factor of safety. Stress value for all the Materials remain same as Stress is function of Load and Area. In this case Pressure load applied and Area of the wing is constant in all the calculations.

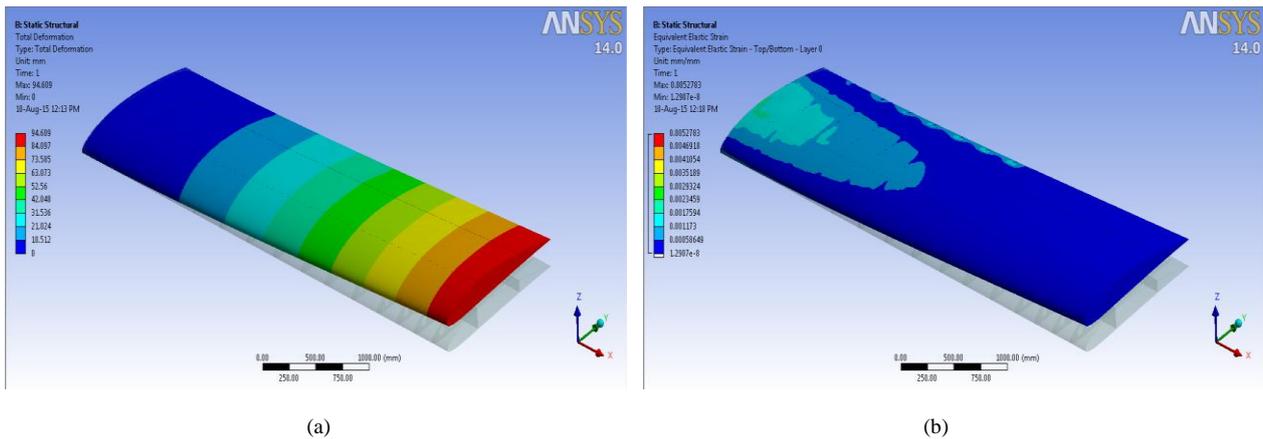
Fig. 6. von-Mises Stress a) Wing with Skin b) Wing without Skin

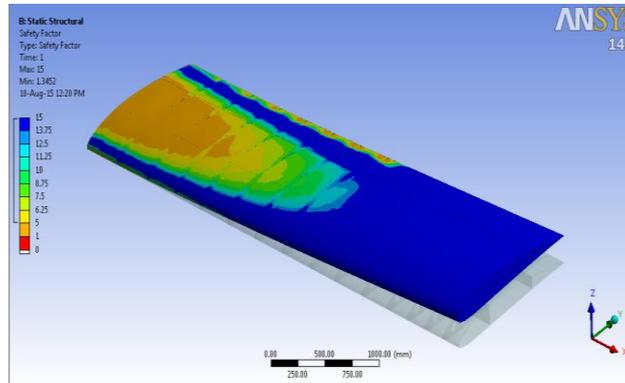


Maximum Equivalent (von-Mises) Stress value obtained for the applied pressure load is 256.46 MPa.

Structural analysis outputs for the Al 2024-T3 wing is as follows.

Fig. 7. Al 2024-T3 wing (a) Deformation (b) Elastic Strain (c) Safety Factor





(c)

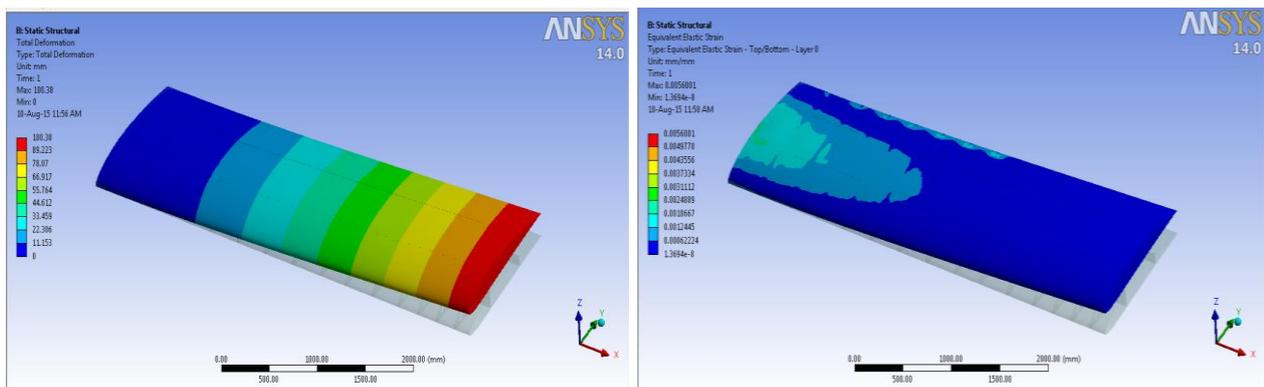
Fig (a) Maximum deformation for the applied pressure load occurs at the wing tip at a value of 94.609 mm.

Fig (b) Maximum Equivalent elastic Strain value obtained for the applied pressure load is 0.00527.

Fig (c) Factor of Safety of the Al 2024-T3 wing is 1.3452 minimum. As the minimum FOS value is greater than 1, the wing can with stand the maximum pressure load which it encounters during the flight.

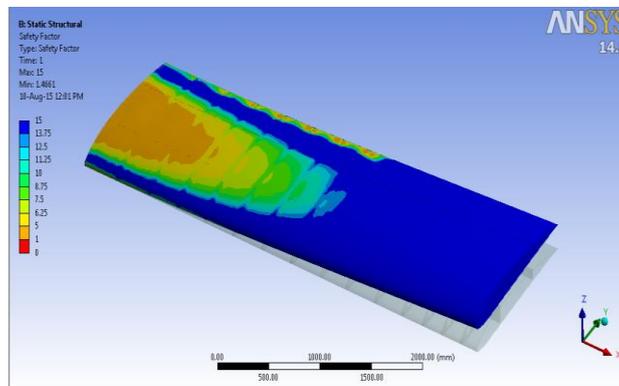
Structural analysis outputs for the Al 6061-T6 alloy wing is as follow.

Fig. 8. Al 6061-T6 wing (a) Deformation (b) Elastic Strain (c) Safety Factor



(a)

(b)



(c)

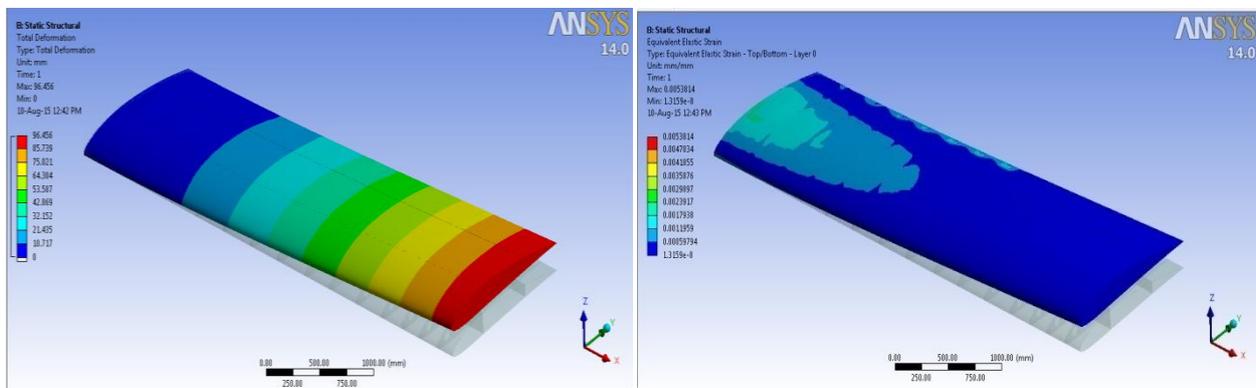
Fig (a) Maximum deformation for the applied pressure load occurs at the wing tip at a value of 100.38 mm

Fig (b) Maximum Equivalent elastic Strain value obtained for the applied pressure load is 0.00560

Fig (c) Factor of Safety of the Al 6061-T6 wing is 1.4661 minimum. As the minimum FOS value is greater than 1, the wing can with stand the maximum pressure load which it encounters during the flight.

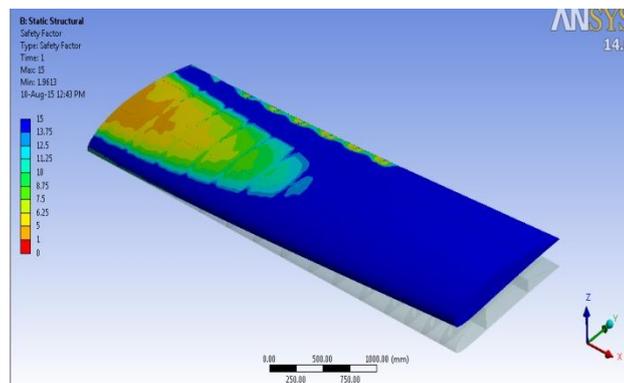
Structural analysis outputs for the Al 7075-T651 alloy wing is as follow.

Fig. 9. Al 7075-T651 wing (a) Deformation (b) Elastic Strain (c) Safety Factor



(a)

(b)



(c)

Fig (a) Maximum deformation for the applied pressure load occurs at the wing tip at a value of 96.456 mm.

Fig (b) Maximum Equivalent elastic Strain value obtained for the applied pressure load is 0.00538.

Fig (c) Factor of Safety of the Al 7075-T651 wing is 1.9613 minimum. As the minimum FOS value is greater than 1 and nearly equal to 2, the wing can with stand the twice the maximum pressure load which it encounters during the flight.

Structural analysis outputs for the Al 7075 + 15% Fly Ash MMC wing is as follows.

Fig. 9. Al 7075 + 15% Fly Ash MMC wing (a) Deformation (b) Elastic Strain (c) Safety Factor

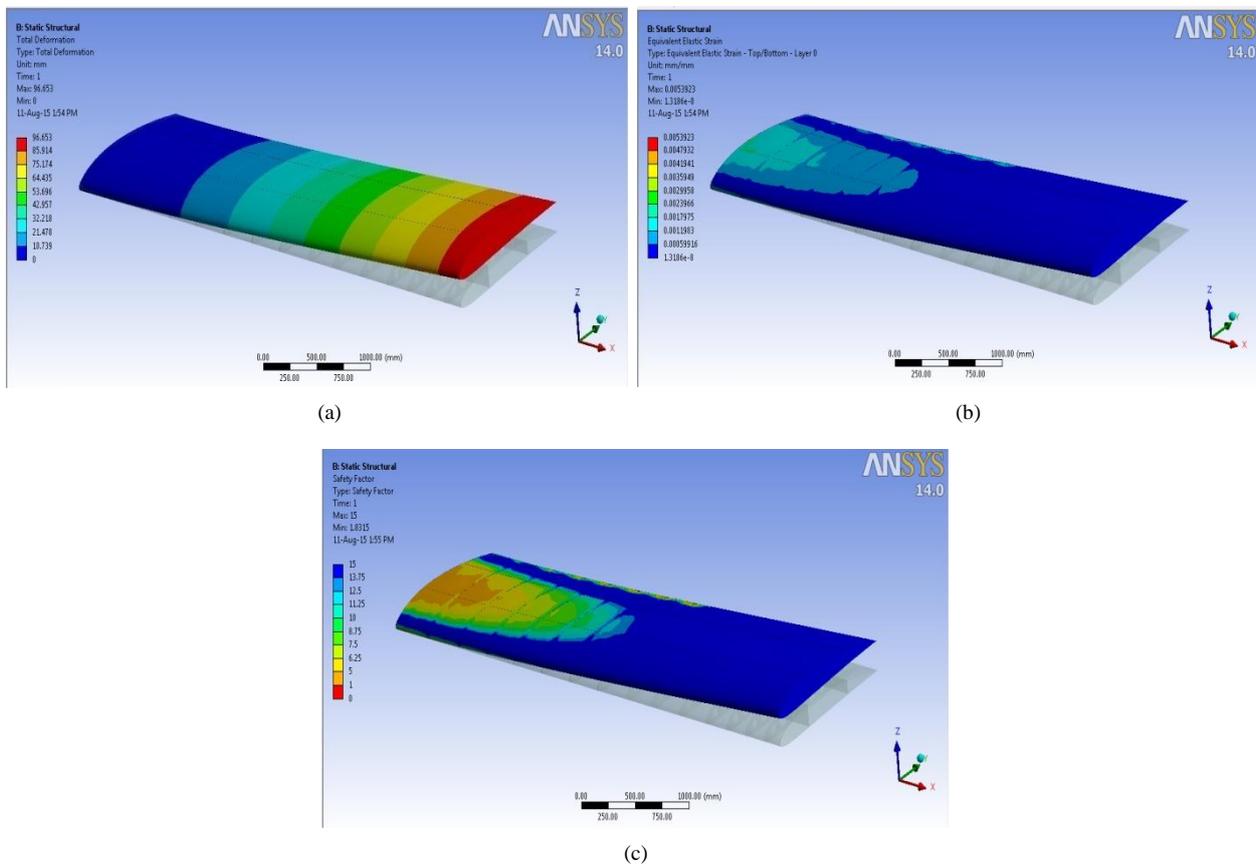


Fig (a) Maximum deformation for the applied pressure load occurs at the wing tip at a value of 96.653 mm..

Fig (b) Maximum Equivalent elastic Strain value obtained for the applied pressure load is 0.00539.

Fig (c) Factor of Safety of the Al 7075+15% Fly Ash MMC wing is 1.8315 minimum. As the minimum FOS value is greater than 1, the wing can with stand the maximum pressure load which it encounters during the flight.

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Table 2 Structural analysis results comparison

Material	Deformation (mm)	Elastic Strain-e	von-Mises Stress (MPa)	Factor of Safety
Al 2024-T3	94.609	0.00539	256.46	1.3452
Al 6061-T6	100.38	0.00560	256.46	1.4661
Al 7075-T651	96.456	0.00538	256.46	1.9613
Al 7075+15%Fly Ash	96.563	0.00539	256.46	1.8315

Pressure load results from the CFD analysis is applied on the wing top and bottom surfaces, structural analysis is done and results are tabulated as shown above Table 2.

V. RESULTS & CONCLUSION

As per the calculated design requirements, the modeling of wing of an ultralight aircraft was done with the help of designing software CATIA V5R20 and the flow analysis was carried out to visualize the flow pattern over the wing section and its behavior like pressure distribution, lift and drag curves were plotted. The structural analysis of the wing section was carried out for materials such as Al 2024-T3, Al 6061-T6, Al 7075-T651 & Al 7075 + 15% FLY ASH MMC with help of ANSYS FLUENT, the results were compared.

From the comparisons we would like to conclude as, Al 7075-T651 material has better structural characteristics than other Alloys. But there are possibilities to use Al 7075 + 15% Fly Ash MMC as there will reduction in weight of the wing as the 15% weight of the Aluminium 7075 is replaced by 15% Fly Ash, which in turn helps in increasing the aerodynamic characteristics as the Drag due to weight is reduced. Even the Factor of Safety (1.8315) for the Metal Matrix Composite is greater than the required value. Margin of Safety=FOS-1 which gives value of 0.8315 is also greater than 0 and nearly equal to 1. As the demand for lighter material with good structural characteristics increases in aerospace and automotive industries, Metal Matrix Composites can be the low cost solution than the Laminated Composite Materials.

As future enhancement, different materials can be tested with same conditions to find more suitable material with good aerodynamic and structural characteristics, number of main load carrying members can be changed and analysis can be performed.

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