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Design and Analysis of Solar Powered RC Aircraft

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ABSTRACT

The main objective of this project is to design and analyze aerodynamic as well as structural loads of the remote controlled aircraft which uses solar energy emerged by the sun as a fuel. Designing of a solar powered aircraft is a challenging task. Because using solar panels to generate enough energy to fly an aircraft is very difficult. The generation of energy depends on the operating geographical area, weather and number of panels used on the aircraft. The wing span of the aircraft should be increased to mount the solar panels to produce more energy. This results in increasing weight and drag, which also increases number of ribs and length of spars. According to these conditions stiffness of the wing has to be increased to prevent from failure. Aluminium and Poly-urithene foams are used for ribs and spar structure. RC aircraft is designed by using Xflr software. 3D modeling is performed using Catia v5. This design is imported to Nastran Patran for analysis where linear analysis is carried out on wing structure. The distribution or variation of aerodynamic loads, maximum principle stress, shear stress and maximum deflection are tabulated after the analysis. Reserve factor values for the ribs, spars and skin of the wing are estimated. Stability and factor of safety are determined using the analysis as per the airworthiness standards.

Keywords *- Angle of attack, principal stress, shear stress, aerodynamic loads, reserve factor, factor of safety.*

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I. Introduction

The success of a solar powered aircraft which can fly continuously was a great challenge a couple of years back, yet this phenomenal test has been able to be conceivable today. To be perfectly honest, discriminating advances have been recognized starting late in the spaces of versatile solar cells, high imperativeness thickness batteries, UAVs are generally used as a piece of military applications for affirmation, regular recognition, ocean observation and mine departure works out. Non-military applications are characteristic surveillance, rice paddy remote recognizing and sprinkling and also system upkeep. There has risen a late criticalness for the aerospace business to discover more practical answers for existing innovations. Blazing plane fuel is one of the numerous supporters to carbon dioxide emanations (among others). A majority of our exploration was devoted to assessing the capability of sun based power as an option force source. Through this examination - and in addition some introductory outline – it was chosen to enlarge a RC model aircraft to outfit solar powered in conjunction with its standard battery framework. The idea is very straightforward, furnished with the solar panels skinning its wing; it recovers vitality from sun keeping in mind the end goal to gives energy to the drive framework and controlling hardware, and then accuses battery of the excess of vitality. Amid in night, the main vitality accessible originates from battery, which releases gradually until the following morning when another cycle begins. Expansion to solar panels includes a dynamo which runs battery power. The impetus cycle begin by first supplying force from solar cells or photovoltaic cells to batteries. Batteries store the charges and supplies to the engine in turn engine runs and it will interface the propeller with a dynamo by certain instrument which thus produces force is put away in batteries.

II. Methodology

Aircraft designing is a process that extends of creating and /or sketching new flight on the paper. These process are classified in to three types and they are

a. Conceptual.
b. Preliminary.
c. Detailed.

The methodology provides conceptual design from which general requirements and size is estimated. These are carried by using preliminary determination of aerodynamics and weight to make a better final unit. The plan's practicality to finish a given mission is set up however the subtle elements of the setup are not characterized. That will likewise consider just steady flight. Where it is planned to accomplish reconnaissance at small elevation or serve as height correspondence stage, sun powered flying machine fit for consistent flight needs to fly at steady elevation. Actually, the first get futile to ground observation at upper elevation and another will not cover an adequate territory at small altitude. For this situation, the vitality and mass parities are the beginning stage of the configuration. Truth be told, the vitality gathered amid the day by the sun powered boards must be adequate to control the engine, the on-load up gadgets furthermore gives power to battery which gives sufficient energy to fly in sunset to the following sunrise when another period begins. In like manner, the lift power needs to adjust precisely the plane weight so that the elevation is kept up.

Tools used

- **a.** XFLR
- **b.** CATIA V5 R19
- **c.** MSC Nastran / Patran

III. Basic Designing Reqirements

The designing of main wing and tails are the basic requirements of an aircraft.

Weight, Lift and Thrust estimation

When a wing/airfoil is moving in a fluid, the fluid will passes through the upper and lower surfaces. As a result fluid/air moves faster on upper surface than the lower surface. By the pressure differences between upper surface and lower surface **Lift** will be generated.

For steady flight,

Weight (W) = Lift (L)

Drag (D) = Thrust (T)

Thrust

Generally need to estimate/fix the speed of an aircraft before estimating its weight. Thrust force is the force required for the plane to move forward. For real flight it should be greater than the Drag force. For this aircraft there is a fixed the speed, which is 16m/s.

Mass & lift

For a flight, it is possible that we can locate the mass of the flight and try to outline the plane with a specific end goal to increase enough lift or concentrate the lift that most likely assemble and try to coordinate the comparing weight. Considered the last, since it is simpler to make a plane lighter than to expand its lift. Making a plane lighter should be possible through uprooting a few bits of hardware or supplanting them by lighter ones (saving money on the battery for instance). Then again, attempting to build lift takes an awesome exertion, in light of the fact that it requires overhauling and assembling the whole wings.

Lift can be calculated by this equation

$L = ½\, C_l \times v^2 \times \rho \times A$

where, L = Lift (Newton)

cl = Coefficient of Lift

ρ = Density of air (kg / m^3)

A = Surface area of wing (m^2)

v = Velocity (m/s)

Angle of attack (α)

The angle between the relative winds with the wing. If the angle of attack rises, as the result lift will also increases but only up to a certain limit point until the airflow would be smooth over the wing separation and the generation of lift cannot be generate. At this time, the sudden dismiss of lift will result in the stalling of the flight, At which the weight of the aircraft no longer supported.

Reynolds number

Other than figuring out whether our wings produce enough lift, we should likewise know whether our wings really create lift by any means. To be specific, tenderly and laminar wind current over a wing will just begin at a sure speed. Moreover, when flying too quickly, the wind current will likewise turn out to be excessively unpredictable and turbulent. The velocity range, at which the wind stream will be smooth and laminar, can be evaluated utilizing a worth called the Reynolds number.

Reynolds number is a worth given to the stream conditions around items. For any article the ideal Reynolds number contrasts, yet as a dependable guideline we can accept the Reynolds number should be between 5.0 × 104 and 2.0 × 105 for airplane wings. The Reynolds number can be computed through the accompanying equation.

Re = v × L × μ-1× ρ
where, Re = Reynolds number
ρ = Fluid density (kg/m^3)
v = Fluid or gas velocity (m/s)
L = Length of fluid or gas travelled (m)
μ = Dynamic viscosity (Pa× s)
Consider the Reynolds number 2.2×10^5
And for steady flight, L=W
So estimated that over all weight of our flight should be within 2.5 Kg
There for, W= 2.5 kg
L=W= 2.5
From literature survey, by thumb rule for RC aircraft **wing loading, weight/wing area (W/S)** is **6 kg/m^2**
Density of air is 1.225 kg/ m^3

$$V = 16 \text{ m/s}$$

$$\text{i.e, } 2.5/S = 6$$

$$S = 0.41 \text{ m}^2$$

$$\mathbf{C_l} = (2\times L)/(\rho \times A \times v^2$$

$$\mathbf{C_l} = (2\times2.5\times9.81)/(1.225\times0.41\times16^2)$$

$$\mathbf{C_l = 0.4}$$

Coefficient of lift for angle of attack (α) 3^0 to 4^0, $\mathbf{C_{l\ 3\ to\ 4^0} = 0.4}$
By the calculated values of the **C_l, S** and **lift** we designed the **main wing** of this aircraft.

A. Airfoil and wing design for small, low-speed air vehicles

A basic introduction to airfoil and wing aerodynamics is discussed below. It includes web-based programs for airfoil design and wing analysis.

Wing design for small UAV's includes some new considerations, however, and this note is meant to highlight some of these.

1) Wing Section Design

Typical wing design for large aircraft is often driven by transonic performance and cruise drag, but for small UAV's may be constrained by the rate of climb or endurance and are often drive by propeller systems that supply approximately fixed output of power rather than fixed thrust. By this changes the appropriate aerodynamic merit function for wings and airfoils. In particular, for maximum range with a given energy, we may wish to maximize L/D, but for a fixed power propulsion system, design for maximum climb rate or endurance requires maximizing C_l 3/2 / C_d. Of course, this does not mean that the airfoil C_l 3/2 / C_d should be maximized, since the drag of the fuselage and tails increases the optimal cl, while wing induced drag tends to lower the value. One needs to include these effects in selecting the best airfoil and design point(s). In any case, it is usual that the overall airplane performance requirements drive the section to operate at its highest practical lift coefficient. A second important consideration in small sized aircraft airfoil design deals with the relatively low Reynolds numbers. In addition to the usual difficulties with flow separation and higher skin friction coefficients, the lower Reynolds numbers makes it harder to achieve the high lift coefficients demanded by long endurance operation. This is compounded by the fact that low Reynolds numbers with higher skin friction increase the optimal design C_l.

Some of the issues important in the design of these low Reynolds number sections are exaggerated when the design leads to insect-scale aerodynamics. The following paper on airfoil design at ultra-low Reynolds numbers contains some ideas relevant to model-airplane sized designs in addition to the "Micro" and "Nano" air vehicles discussed there. Kunz, P., Kroo, I., "Analysis, Design, and Testing of Airfoils for Use at Ultra-Low Reynolds Numbers", In Fixed, Flapping and Rotary Wing Aerodynamics for Micro Aerial Vehicle Applications, T. Meuller, ed., AIAA.

2) Wing Design

Again, many of the considerations related to lift and C_l distributions for large aircraft wing design are relevant here. The basic trade between aerodynamics and structures affects the choice of wing aspect ratio however; the wing structural weight varies rather differently from large aircraft structures, leading to different platforms. In addition, the fact that higher aspect ratio leads to lower Reynolds number and higher skin friction drag coefficient lead to lower optimal aspect ratios. The optimal aspect ratio can be quite small at very low Reynolds numbers, even when structural effects are excluded. Finally, the fact that many of these designs operate near maximum C_l, means that it is especially important to assure acceptable stalling characteristics. This often leads to larger tip chords than would be selected otherwise. The deleterious effect of lower Re on section maximum C_l also pushes these designs to larger tip chords (higher taper ratios) than would be optimal for large aircraft.

Airfoil selection

Geometry of an Airfoil can be characterized by the coordinates of the upper and lower surface. It is then explained by a few parameters like, maximum thickness of airfoil, maximum camber of airfoil, position of max thickness in an airfoil, position of max camber in an airfoil, and the radius of nose. So anyone can generate an airfoil section by the given parameters as above. It's done by Eastman Jacobs in the early of 1930's to make a series of airfoils called the NACA Sections.

Nomenclature of an Airfoil:

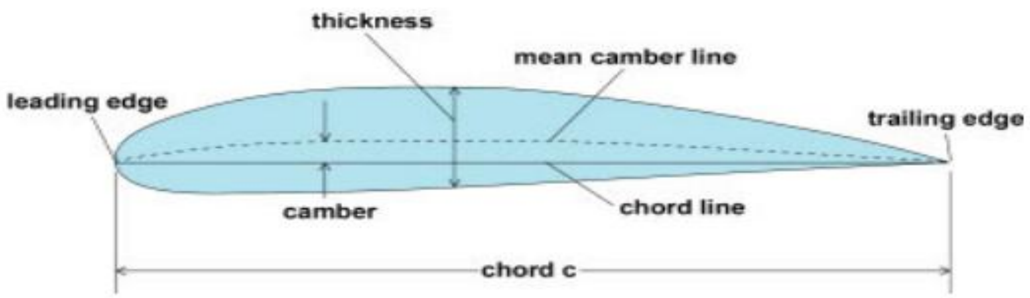


Fig nomenclature of Airfoil

- **a)** c, is the chord length which is the length from the LE to the TE of a wing cross section which is parallel to the vertical axis of symmetry of the airfoil.
- **b)** Mean camber line. which line is half between the upper surface and lower surfaces
  - leading edge (LE) is the front point on the mean camber line, trailing edge (TE) is the back point on mean camber line of the airfoil.
- **c)** Camber is the maximum distance between the mean camber line and the chord line, when measured perpendicular to the chord line of the airfoil.
  - 0 camber or un cambered refers the airfoil is symmetric above and below the chord line.
- **d)** Thickness is the distance between upper and lower surface when measured perpendicular to the mean camber line of the airfoil.

Types of airfoil

The NACA 4 digit and 5 digit airfoils were created by superimposing a simple mean line shape with a thickness distribution that was obtained by fitting a couple of popular airfoils of the time

$$\pm y = (t/0.2)\times(0.2969 \times x^{0.5} - 0.126 \times x - 0.3537 \times x^2 + 0.2843 \times x^3 - 0.1015 \times x^4)$$

The camber line of 4-digit sections was defined as a parabola from the leading edge to the position of maximum camber, then another parabola back to the trailing edge.

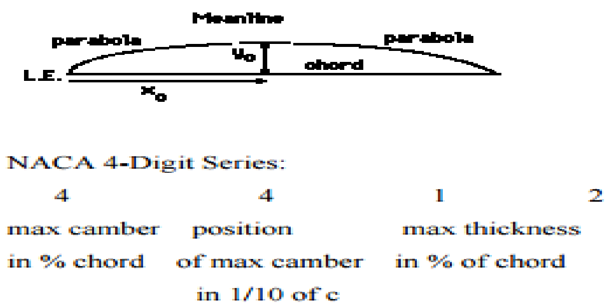


Fig NACA 4-digit series

Main Wing details

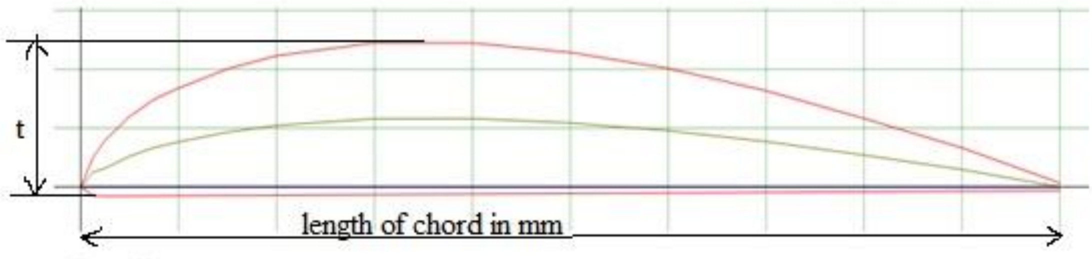
For low speed aircraft generally flat bottom airfoils are used. Because flat bottom airfoil provides low drag with high lift and if the power fails then it helps to glide the aircraft for some times in air. This helps the low speed aircraft to safe land.

Comparison of airfoils for main wing

Compared these airfoils with each other. Comparison was based on **angle of attack**, **C_l/C_d, C_l/α, C_d/α and C_m/α**. So by the smooth curve got from the graph so finalized **GOE 611** airfoil

1) CLARCK-Y AIRFOIL
2) GOE 446 AIRFOIL
3) GOE 528 AIRFOIL
4) GOE 412 AIRFOIL
5) USA 27 mod. AIRFOIL

Selected **GOE 611** airfoil because of the reason it is most commonly used in gliders and which is easy to fabricate by hand.



Details

(goe611-il) GOE 611 AIRFOIL
Gottingen 611 airfoil
Max thickness 12.9% at 29.9% chord.
Max camber 5.8% at 39.9% chord

Fig airfoil of main wing

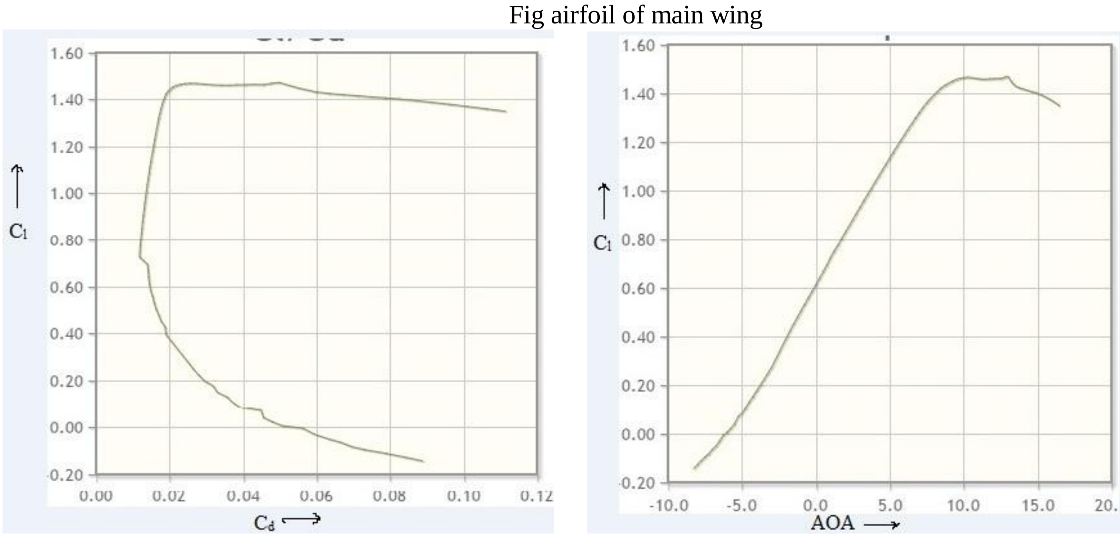


Fig a: graph comparison

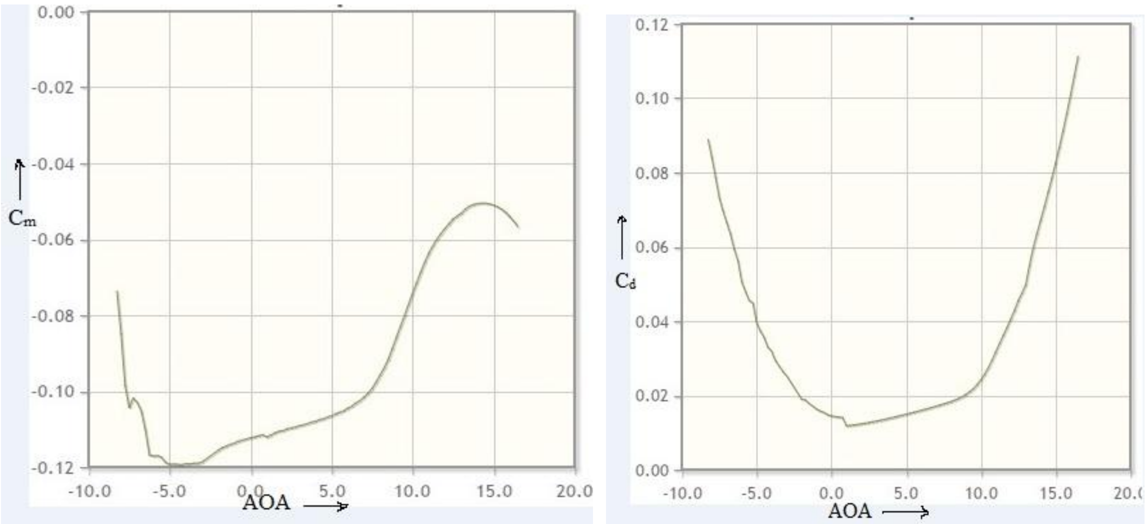


Fig b: graph comparison

The point in airfoil where the lift can be supposed to be concentrated upon is called the center of pressure.

- Commonly it is located at c/4 of the airfoil (where c is the chord length).
- The point at which the weight of the aircraft acts is termed as center of gravity (CG).
- The CG must coincide with the center of pressure for weight balance in the aircraft.
- To locate CG to c/4, at the nose can add some weight (like paper clips, small coins).

Tail plane details

Tail plane is an aerodynamic surface used to stabilize and control the longitudinal (pitching) and/or directional (yawing) moment of an aircraft.

For the tail plane, we have two stabilizers. One is horizontal stabilizer where elevator is mounted (for pitching moment) and the other is vertical stabilizer where the rudder (for yawing moment) is mounted.

Some of the thumb rules considered for tail plane are

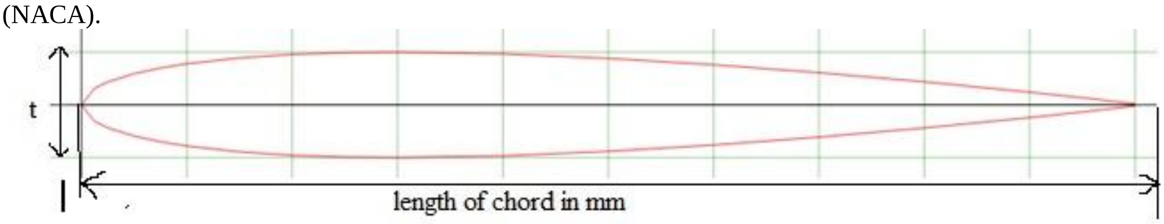
- wing span of horizontal tail will be $1/4^{th}$ of the span the main wing
- wing span of the vertical tail will be $1/2^{th}$ of the horizontal tail
- area of the horizontal tail will be equal to 15 to 20% of the main wing area
- horizontal stabilizer should be mount at the distance of (2 to 3) times of length of the main wing chord

Airfoil selection for tail (both horizontal and vertical stabilizer)

Compared these airfoils with each other. Comparison was based on **angle of attack**, **C_l/C_d, C_l/α, C_d/α and C_m/α**. So by the smooth curve got from the graph so finalized **NACA 0010** airfoil.

1)NACA 0010-34
2)NACA 0010-35
3)NACA 0010-64
4)NACA 0012
5)NACA0015

4 digit NACA airfoil i.e., NACA 0010, where camber is 0 and to chord length proportion is 10% thickness. NACA airfoils are outer figure/sketch for wing of airplane. National Advisory Committee for Aeronautics (NACA).



Details

(naca0010-il) NACA 0010
NACA 0010 airfoil
Max thickness 10% at 30% chord.
Max camber 0% at 0% chord

Fig 4.5.1: airfoil of tail

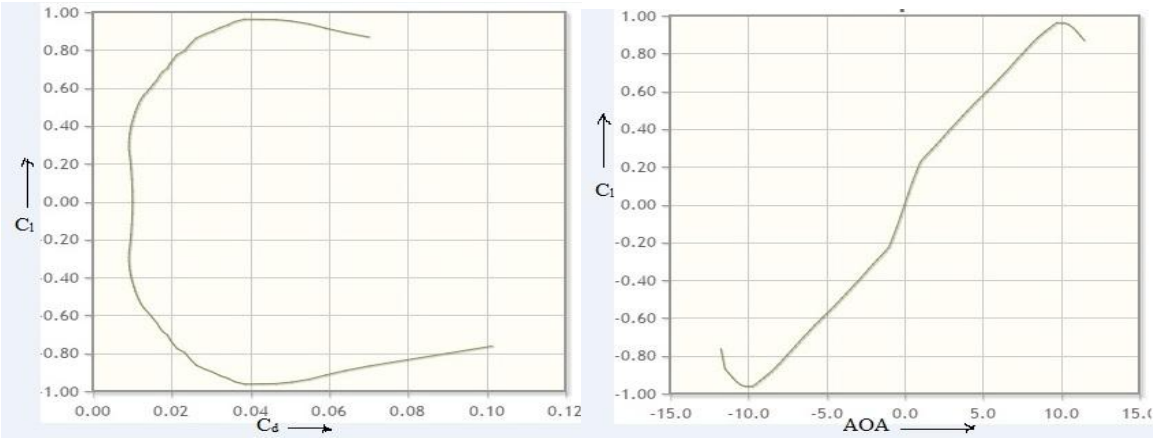


Fig a: graph comparison

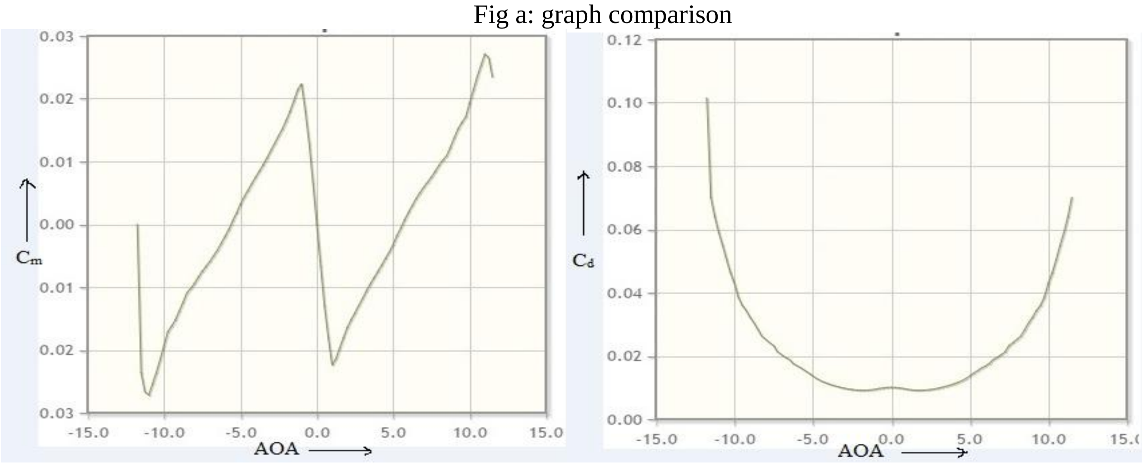


Fig b: graph comparison

Tail volume coefficient

Tail volume coefficient = area of the wing/ area of the tail, have the value 0.7 (generally)

i.e., **v= $(L_t \times S_t) / (S_w \times C_w)$**

where, $\mathbf{L_t}$ is the distance from AC of wing to CG of the tail

$\mathbf{S_t}$ is the area of tail

$\mathbf{S_w}$ is the area of wing

$\mathbf{C_w}$ is the wing chord length

XFLR Model of Aircraft Analysis.

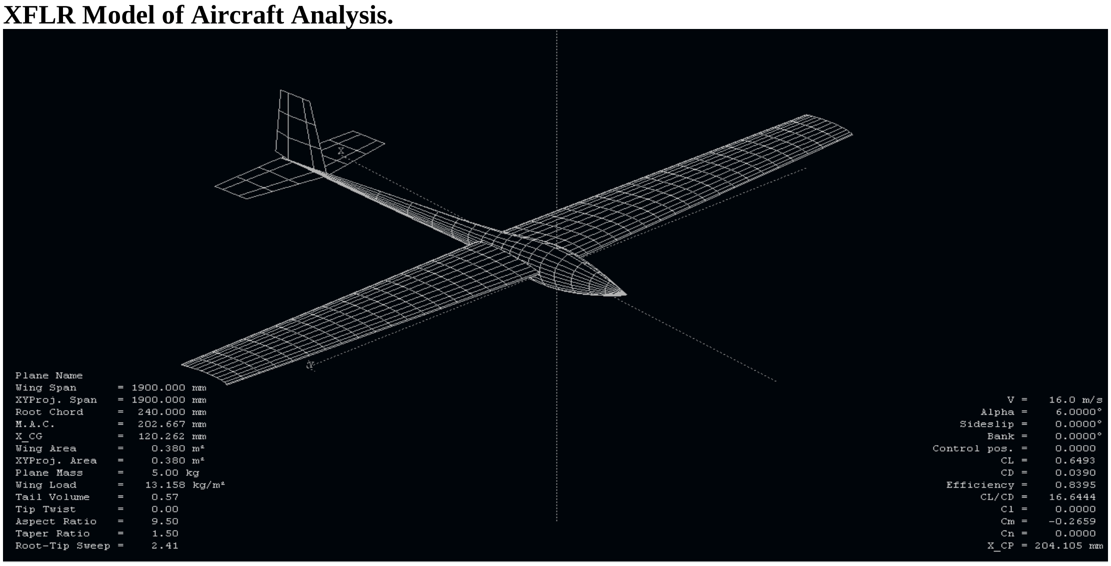


Fig XFLR model of Aircraft Analysis

The above figure shows the designing and specification of our solar powered rc aircraft.

SPECIFICATIONS	WING	HORIZONTAL TAIL	VERTICAL TAIL
Wing span	1900 mm	420 mm	400 mm
Root chord	240 mm	210 mm	180 mm
Tip chord	160 mm	110 mm	100 mm
Area	0.38 m^2	0.06 m^2	0.03 m^2
Mean aero chord	202 mm	136.54	143.81
Aspect ratio	9.50	3.11	2.86
Root to tip sweep	2.41^0	7.46^0	5.71^0

Table Specifications of wing and tail

Design specification for the Solar powered RC Aircraft

Parameter used	Value
Wing Span	1900 mm
Root Chord length	240 mm
Airfoil type	GOE611
Mass of the structure	0.5 kg
Wing Load	13.15kg/m^3

Table Design specifications

Catia Model of XFLR Design

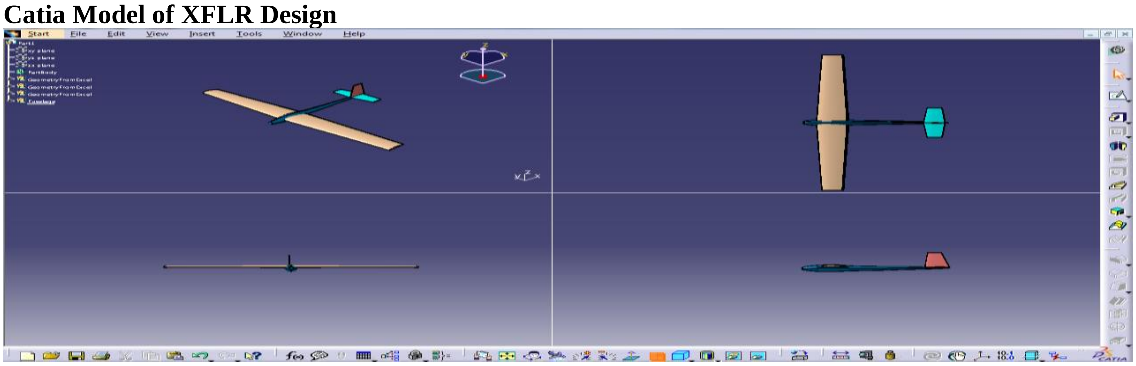


Fig Catia model of XFLR design

Above figure shows the front, top, side and isometric view of the solar powered aircraft

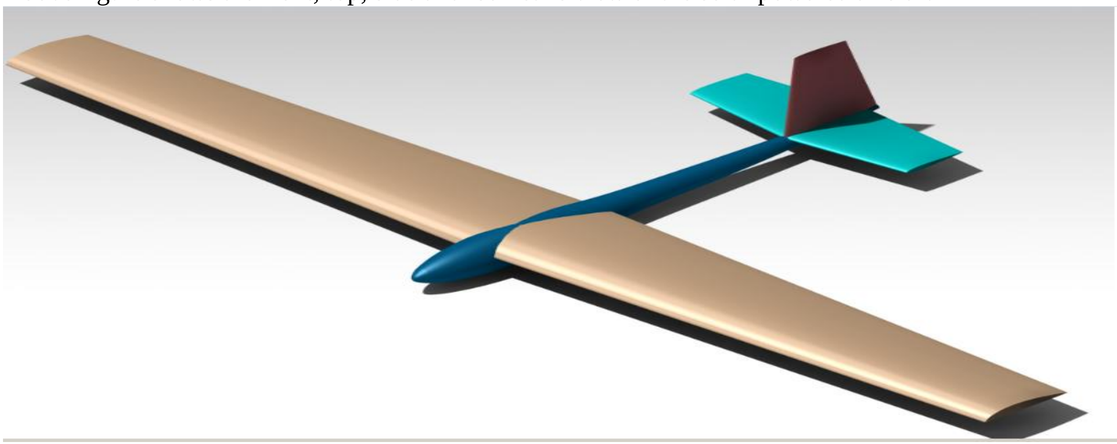


Fig 5.3.6.1: Isometric model

IV. STRESS ANALYSIS

Importing the 3d Model of the wing

The 3d model of the wing which was designed in Catia is imported to Nastran and Patran for the Stress Analysis.

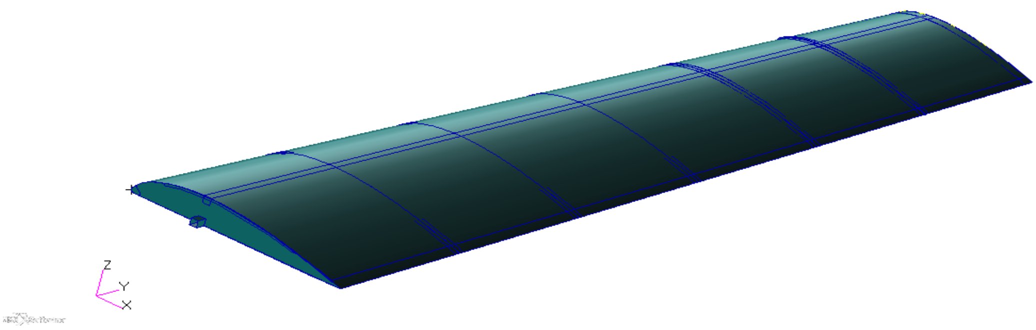


Fig Imported model isometric view

Discretization of the Structure

The next step in the finite element method is to divide the structure region into subdivisions or elements. Hence, the structure is to be modeled with suitable finite elements. The number, type, size, and arrangement of the elements are to be decided. Next is verification of mesh elements. First verified for element boundary, duplicate elements, normal's and geometry fit meshing Quad element is checked for error. Errors in quad element are

- Aspect ratio
- Warp
- Taper
- Skew

If there is any error rectify the errors by processing modifying mesh.

Finite Element Modeling

Sl.no	Element Type	Number of elements
1	Bar	242
2	Quad	3039
3	Tria	96

Table Type of Elements Used

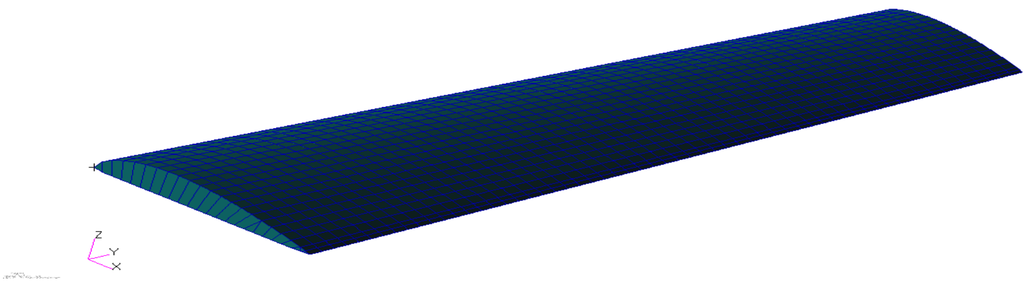


Fig Finite Element Model

Material Properties

Selection of aircraft materials depends on many considerations, in general can be categorized as cost and structural performance. Cost includes initial material cost, manufacturing cost and maintenance cost. The key material properties that are important to maintenance cost and structural performance are

- Density (weight)
- Stiffness (young's modulus)
- Strength (ultimate and yield strengths)
- Durability (fatigue)
- Damage tolerance (fracture toughness and crack growth)
- Corrosion

Element Properties

SL.NO	Element Type	Component	Properties
1	1D	Front spar	Hollow Square (w=8,h=8,t=1) L-Section(w=8,h=8,t=1)
		Nose spar	W=12;H=12: Material: Aluminium
2	2D	Ribs	t = 13
3	2D	Skin	t = 3

Table Finite Element Model Properties

Where, w- Width; h-Height; t-thickness (all in mm)

Applying LOADS and BOUNDARY CONDITIONS

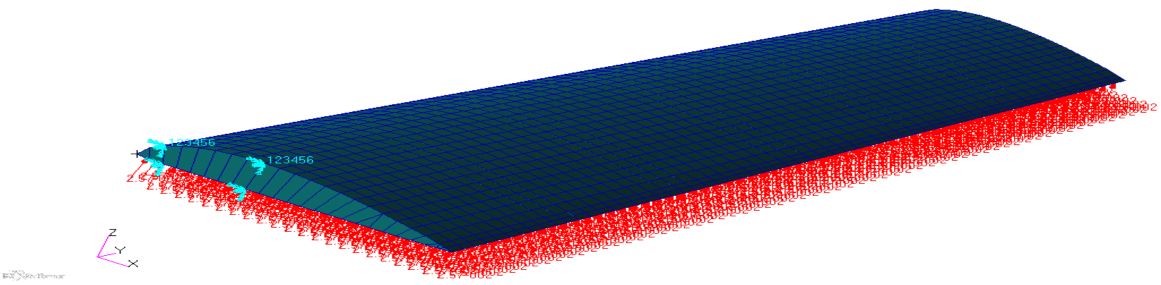


Fig Boundary Conditions

V. ANALYSIS

After the application of Loads and Boundary conditions, with all the inputs of material selection and definition of properties, the FE model of the component is ready to undergo the analysis procedures. The type of analysis chosen here is a Linear static analysis with solution sequence code SOL101, the BDF (Bulk data file) is created, it contains information about elements grid or node spc and material properties load and boundary conditions, and BDF is the input file for the Solver (Msc.Nastran). The bdf of composite fuselage is shown below.

From the linear static stress analysis, from the Nastran &Patran software, the Von mises stress, Maximum principal stress, Minimum Principal Stress, Maximum Shear Stress and the displacement results are obtained and those results are quick plotted in Patran and simulated for better understanding of the behavior of the structure for the loads applied.

VI. RESULTS

Reserve Factor

The result validation is done based on material ultimate strength with R.F (**Reserve Factor**) table.

R.F= ultimate strength/obtained stress

Note:

- If R.F value is less than 1 means component fails.
- If R.F value is greater than 1 to 1.5 means component is safe.
- If R.F value is greater than 1.5 means component is too safe.

Calculations

Maximum take of wieght = 3.5 kg× 9.81 = 34.335 N

Limit load = take of wieght × lift

Load factor at max “G” condition (lift load) = 1.5 G

34.335× 1.5 = 51.5025 N

Design load = limit load × factor of safety

Factor of safety = 1.5

51.5025× 1.5 = 77.2537

Load of semi span = design load/ 2
77.2537/ 2 = 38.627
Pressure load on wing = load on semi wing span / wing area
38.627 /1500 = 0.0257 N/mm^2

Reserve factor table

Sl No.	Component	Types of stress	Result obtained	Units	Ultimate tensile strength of material	Reserve factor
1	Skin	Von mises	19.5	N/mm^2	20.7	1.061538
		Max Principal	18.7	N/mm^2	20.7	1.106952
		Max Shear Stress	9.98	N/mm^2	12.42	1.244489
2	Ribs	Von mises	19.5	N/mm^2	20.7	1.061538
		Max Principal	18.7	N/mm^2	20.7	1.106952
		Max Shear Stress	9.98	N/mm^2	12.42	1.244489
3	Nose Spar	Von mises	1.38	N/mm^2	480	347.8261
		Max Principal	1.56	N/mm^2	480	3076923
		Max Shear Stress	0.784	N/mm^2	480	614.5967
4	Front Spar	Von mises	10.8	N/mm^2	480	44.44444
		Max Principal	10.2	N/mm^2	480	47.05882
		Max Shear Stress	5.74	N/mm^2	480	83.62369

Table Reserve factor table

FEA result table

Sl. no.	Component	Type of stress	Result Obtained	Material
1	Skin	Von mises (N/mm^2)	19.5	PU-Foam
		Maximum Principle stress (N/mm^2)	18.7	
		Maximum shear stress (N/mm^2)	9.98	
		Displacement(mm)	56.7	
2	Ribs	Von mises (N/mm^2)	19.5	PU-Foam
		Maximum Principle stress (N/mm^2)	18.7	
		Maximum shear stress (N/mm^2)	9.98	
		Displacement(mm)	11.2	
3	Nose spar	Von mises (N/mm^2)	1.38	Aluminium
		Maximum Principle stress (N/mm^2)	1.56	
		Maximum shear stress (N/mm^2)	0.781	
		Displacement(mm)	0.00947	
4	Front spar	Von mises (N/mm^2)	10.8	Aluminium
		Maximum Principle stress (N/mm^2)	10.2	
		Maximum shear stress (N/mm^2)	5.74	
		Displacement(mm)	0.041	

Table FEA result table

FEA results for skin (pu-foam material)

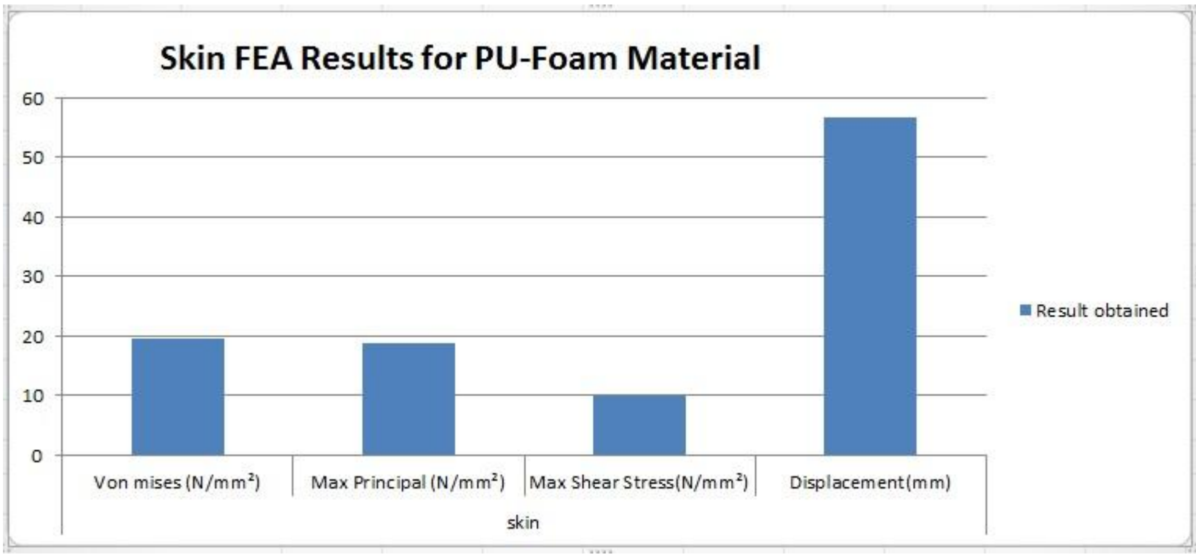


Fig FEA results for skin

FEA results for ribs (pu-foam material)

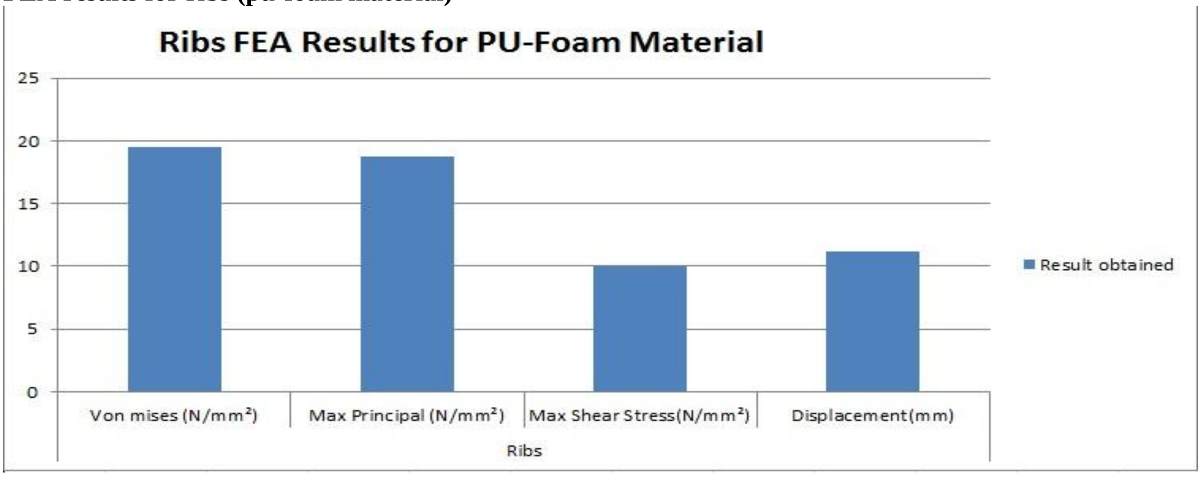


Fig FEA results for ribs

VII. CONCLUSION

- Design of the aircraft which includes aerodynamic forces (lift is 2.5 and thrust 16m/s) are calculated and C_l/C_d, C_l/AOA, C_d/AOA and C_m/AOA graphs are also plotted using Xflr software by considering the GOE 611 airfoil for main wing and NACA 0010 airfoil for tail.

- Designed model from the Xflr software has designed in 3D model using Catia v5 software by considering the ribs and spar structure of wing which includes front and nose spar with 10 numbers of ribs.

- Analysis of the main wing has been carried out using Nastran and Patran software where von mises stress, maximum principal stress, maximum shear stress and deflection of the spars, skin and ribs of the main wing are calculated by considering skin and ribs are PU foam materials and the spars are Aluminium materials.

- The von mises stress and the Reserve factor of the skin are 19.5 N/mm^2 and 1.061538, ribs are 19.5 N/mm^2 and 1.061538, nose spar are 1.38 N/mm^2 and 347.8261 and for front spar are 10.8 N/mm^2 and 44.4444.

- The maximum principal stress and Reserve factor of skin are 18.7 N/mm^2 and 1.106952, ribs are 18.7 N/mm^2 and 1.106952, nose spar are 1.56 N/mm^2 and 307.6923 and front spar are 10.2 N/mm^2 and 47.05882.

- The maximum shear stress and Reserve factor of skin are 9.98 N/mm^2 and 1.24448, ribs are 9.98 N/mm^2 and 1.24448, nose spar are 0.781 N/mm^2 and 614.5967 and front spar are 5.74 is N/mm^2 and 83.6236.

- The maximum displacement of skin, ribs, nose spar and front spar 56.7 mm, 11.2 mm, 0.00947 mm and 0.041 mm respectively.

The analysis results show that the material is withstanding the applied loads and has deformed under the limits. Hence structure is found to be stiff and safe under the given operating conditions the results are well within the airworthiness standards.

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