

Design Optimization And Analysis Of Airfoil Sections For UAV's

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Abstract: Recent advancements in communications, materials technology and fabrication technology tends to improvisation in design and structural ability of economical UAV's to serve as a multipurpose utility tool in different sectors viz., scientific, civil, agriculture and military sectors. The paper describes the design and structural analysis of a fixed wing UAV for a selected airfoil profile. The paper also describes the conditions and constraints assumed during the design of an UAV and the importance of structural analysis. The design is done in CATIA V5 and structural analysis using ANSYS workbench. Individual parts are designed and are then assembled as per the constraints involved. The structural analysis is done considering the wing to be made of depron foam and assuming the wing to be a cantilever beam.

Keywords: Design, Structural analysis, CATIA V5, ANSYS workbench, Cantilever beam.

1. INTRODUCTION

The selected airfoil for design was CLARKY. Root and tip chord lengths was calculated and were multiplied with the relevant scale factor. With suitable assumptions aforementioned, wing calculations were made and this gave the completed wing part for the assembly.

For the fuselage model it's decided to use a standard aerodynamic design used in flying wings. This design was obtained as an assembly of panels which were individually drafted as 2 dimensional sketches. The fuselage design was made with considering it would contribute to improve aerodynamic stability with minimal drag. To this standard part, a volume section was subtracted using Boolean subtraction at the bottom which would serve as our camera housing. On the top of the fuselage a panel was added using materials options to indicate a removable panel which could be used to add and remove electrical parts of the plane.

Thumb rules for plane design

- The ratio of the wing span to wing root chord should be 5 or 6
- The wing thickness should be 12% to 14% of the wing root chord: • Example: If the wing root chord is 6" then the widest part of the wing should be 3/4" thick. Note: Foam profile planes do not follow this rule of thumb but still fly.
- The fuselage length should be 70% - 75% of the wing span
- The distance from the leading edge of the wing to the back of the prop should be 15% of the wingspan
- The leading edge of the wing to the stabilizer should be 3 times the wing root chord: •
- The horizontal stabilizer should be 25% of the wing area:
- The vertical stabilizer should be 10% of the wing area
- The plane should balance at 25% - 33% of the wing root chord: • Example: If the wing root chord is 6" from the leading edge to the trailing edge of the wing then the Center of Gravity (COG) should be located 1.5" - 2" from the leading edge of the wing

1.1. Equations and Indentations

Wing calculations:

1. Span (b) = 900 mm
2. Root chord = 150 mm
3. Tip chord = 100mm
4. Average chord = $\frac{\text{root chord} + \text{tip chord}}{2} = \frac{100 + 150}{2} = 125$ mm
5. Aspect ratio = $\frac{b^2}{s} = \frac{900^2}{112500} = 7.2$
6. Wing area (s) = wing span × average chord = 900 × 125 = 1,12,500 mm²

Tail wing calculations:

7. A = root chord
8. $B = \frac{LENGTH}{2}$
9. C = tip chord
10. Area of tail wing = $B \times \frac{A+C}{2}$

Horizontal stabilizer:

11. Scale 1:30
12. Area = $180 \times \frac{100+50}{2} = 13500$
13. Aspect ratio = $\frac{360^2}{13500} = 9.6$

Vertical stabilizer:

14. Area = $60 \times \frac{100+150}{2} = 4500 \text{ mm}^2$
15. Sweep angle = 60°

Fuselage calculations:

16. Centre of gravity = located at 25% of mean aerodynamic chord
17. Fuselage length = 75% of wing span = $900 \times 0.75 = 675 \text{ mm}$
18. Leading edge of wing to propeller = 15% of wing span = $0.15 \times 900 = 135 \text{ mm}$
19. Fuselage diameter = 3 times the thickness of chord at the root = $19.97 \times 3 = 59.91 \text{ mm}$
20. Area of horizontal stab in tail = 25% of wing area = $112500 \times 0.25 = 27000 \text{ mm}^2$ (considering both wings)
21. Area of vertical stabilizer = 1/3 of horizontal stabilizer = 4500 mm^2
22. Trailing edge of wing to tail wing = 3 times the MAC = $3 \times 125 = 375 \text{ mm}$

1.2. Abbreviations and Symbols used in Equations

- C_L - Lift coefficient
- C_D - Drag coefficient
- C_M - Moment coefficient
- ρ - Density
- μ - Viscosity
- V - velocity
- S - Reference area
- L - lift force
- D - drag force
- q - dynamic pressure
- b - wing span
- L - length

1.3 Airfoil Structure (2-D and 3-D CAD models)



Fig. 1 Airfoil profile

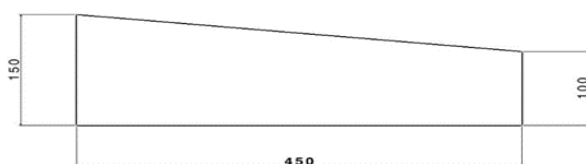


Fig 2. 2-Dimensional Sketch of Wing

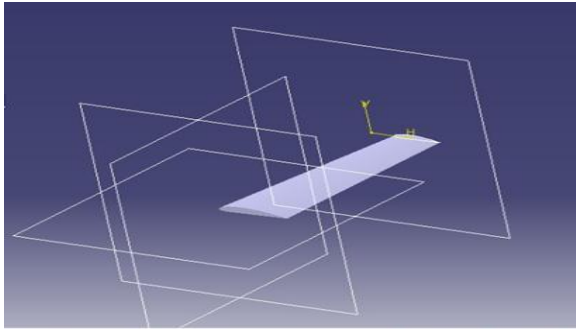


Fig 3. 3-Dimensional design of Wing using CATIA

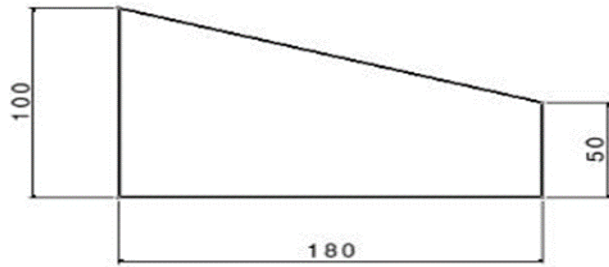


Fig 4. 2-Dimensional design of horizontal stabilizer

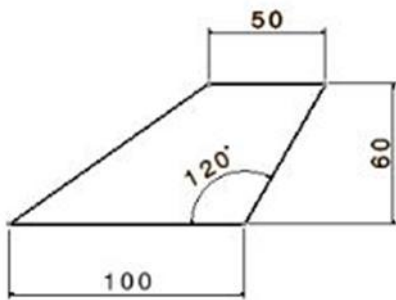


Fig 5. 2-Dimensional design of vertical stabilizer

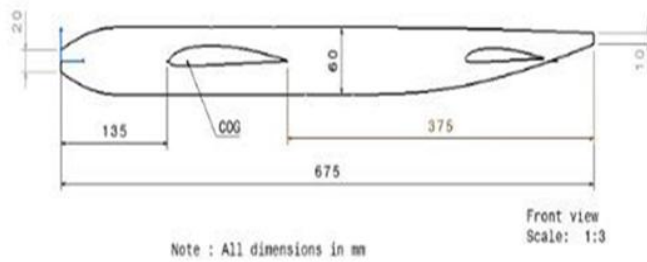


Fig 6. 2-Dimensional design of fuselage

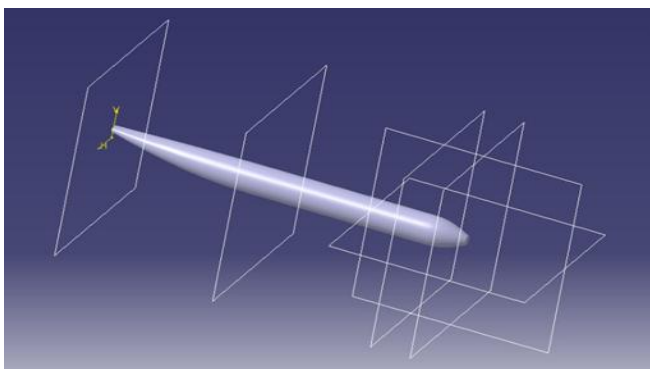


Fig 7. 3-Dimensional design of fuselage using CATIA

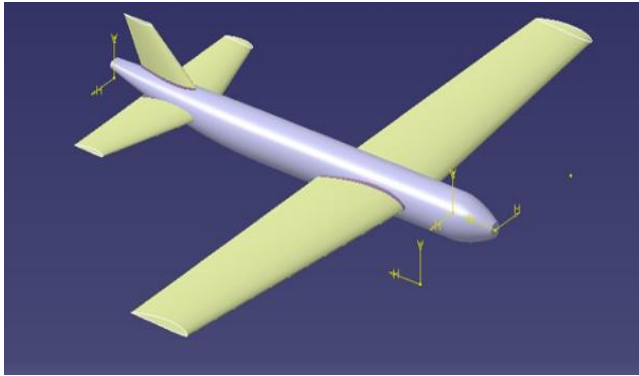


Fig 8. Final Assembled model of UAV

1.4 Structural Analysis

1.4.1 Pre-Processing of Airfoil structure

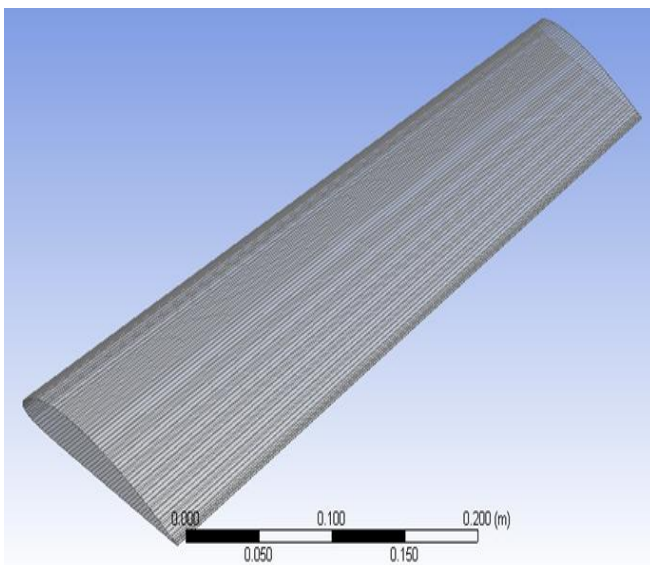


Fig 9. Wing section for structural analysis.

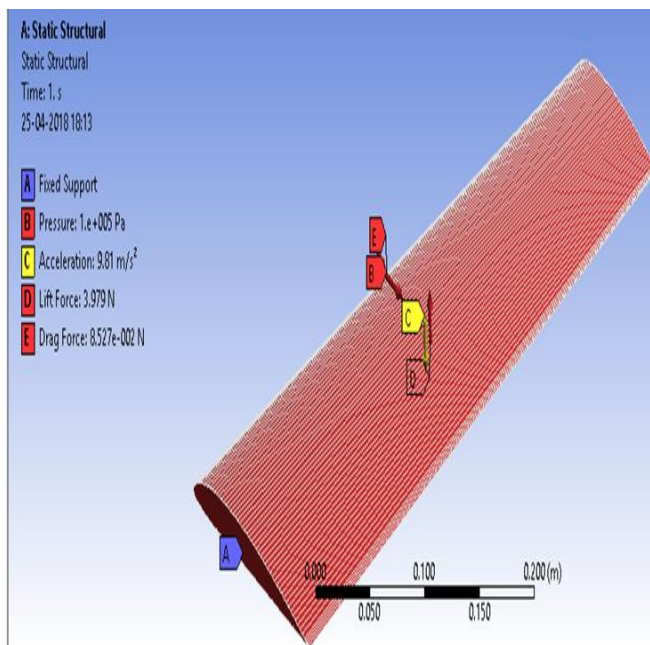


Fig 10. Wing section with pressure and load applied.

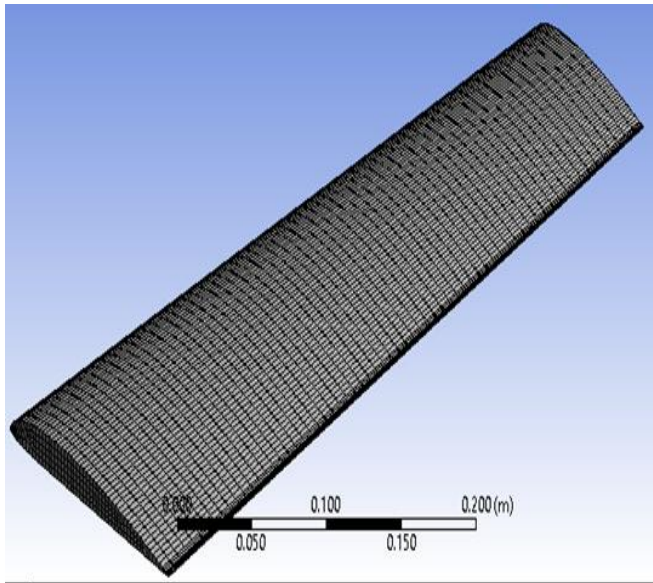


Fig 11. Meshed Wing section.

1.4.2 Post-Processing of Airfoil structure

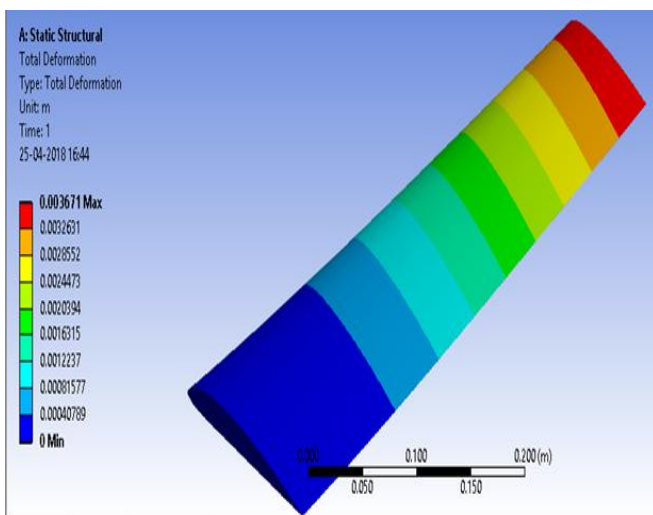


Fig 12. Total deformation of wing structure.

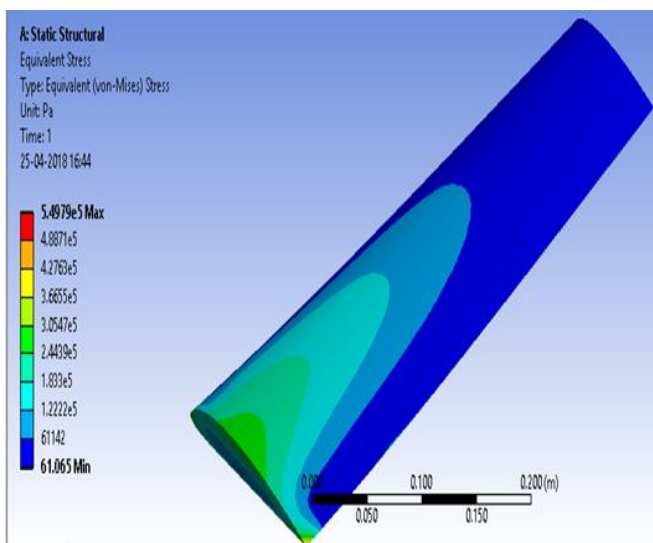


Fig 13. Equivalent Von-Mises stress.

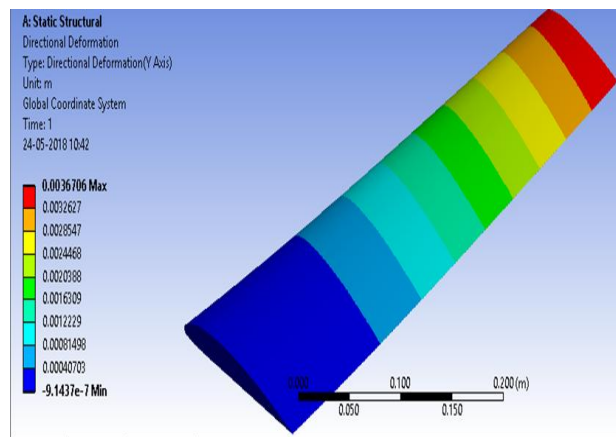


Fig 14. Directional deformation in Y-Axis.

The table of material data for depron which is considered for design and the values obtained from structural analysis is shown

Table 1. Properties of depron material

Property	Values
Density	40 Kg/m ³
Young's Modulus	470 Mpa
Poisson's ratio	0.03
Bulk Modulus	166.67 Mpa
Shear Modulus	228.16 Mpa
Tensile Yield Strength	0.87 Mpa
Tensile Ultimate Strength	14.1 Mpa

1.4 Experimental Results and discussion

Table 2. Deformation and Stress analysis results

Objective			
	Total deformation	Directional deformation	Equivalent stress
State	Solved	solved	Solved
Scoping method	Geometry selection	Geometry selection	Geometry selection
Shell	Top/bottom	Top/bottom	Top/bottom
Layer	Entire section	Entire section	Entire section
Definition			
	Total deformation	Directional deformation	Equivalent stress(von-mises)
By	Time	Time	Time
Display time	Last	Last	Last
Calculate time history	Yes	Yes	Yes
Suppressed	No	No	No
Orientation	Y axis	Y axis	Y axis
Coordinate system	Global coordinate system	Global coordinate system	Global coordinate system
Results			

	Total deformation	Directional deformation	Equivalent stress
Minimum	0 mm	-9.143×10e-4mm	6.1065×10e-5MPa
Maximum	3.671 mm	3.6707mm	0.54979MPa
Minimum value over time			
Minimum	0 mm	-9.143×10e-4mm	6.1065×10e-5MPa
Maximum	0mm	-9.143×10e-4mm	6.1065×10e-5MPa
Maximum value over time			
Minimum	3.671 mm	3.6706 mm	0.54979MPa
Maximum	3.671 mm	3.6706 mm	0.54979MPa

CONCLUSION

The following results and conclusions were obtained based on the structural analysis of the airfoil structure:

- i. The “depron” material for the construction proves to be cost effective when compared to other alloy materials and also depicts good strength to weight ratio.
- ii. The model was successfully designed and analysis was carried out using SHELL element and GEOMETRY SELECTION method for the entire cross section.
- iii. The total deformation in Y direction is almost negligible even after the pressure is applied over a period of time which proves good drag and lift capacity of the selected airfoil section.
- iv. For the maximum total deformation of 3.671mm the Von mises stress was observed to be 0.54979 MPa, which is well below the yield strength of the material (0.87 MPa).

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