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SELECTION AND ANALYSIS OF AN AIRFOIL FOR FIXED WING MICRO UNMANNED AERIAL VEHICLE

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Abstract - *Airfoil plays a vital role in wing designing. The airfoil selection for the wing of unmanned aerial vehicles (UAV) helps better performance to lift. Different airfoil shapes such as symmetrical, cambered, and semi-symmetrical show different natures in aerodynamic & geometrical characteristics. This research deals with the selection of airfoil for a fixed-wing of micro UAV which has a lift capacity of 2.5 kg. Over 2000 airfoils of NACA, MH, G, S, E series are analyzed with aerodynamic characteristics i.e. coefficient of lift & drag (Cl & Cd), coefficient of pressure (Cp), coefficient of the moment (Cm), angle of attack (α), camber & thickness. The airfoil selection is based on better lifting capacity reducing drag. Airfoils for the selection process were available from the airfoil tools database and XFLR 5 software used for analysis. For better accuracy, 150 coordinates were selected for a curve of the airfoil. Modeling and analysis are done in Solidworks & ANSYS software resp. The best suitable airfoil is nominated by analyzing graphs and verified with an analysis model*.

Key Words: **Aerodynamic characteristics, Airfoils, Fixed wing, UAV, XFLR.**

1. INTRODUCTION

An airfoil is a cross-section of the wing and its function is to produce a lift to the plane. Lift is directly proportional to the weight, the force acts as opposed to lifting called drag which resists the motion of the plane. The amount of weight lifted by plane depends on its application and according to that airfoil decided. Nowadays, UAVs use increases in transportation, agriculture, military, and others. Airfoil selection for a UAV wing is the key step of wing designing to perform its function [1]. Wing consists of numbers of airfoils are joined on spar with appropriate distance in them. Proper selection of airfoil is important as it affects the performance of wing & aircraft [2].

This work presents different characteristics, conditions, criteria and processes to be considered while selecting an airfoil. The speed approximately defined for the cruise is about 15 m/s and stall about 10 m/s, weight is 3 kg and air density 1.225 kg/m^3. Different airfoils have been analyzed and the best between them is selected by comparing aerodynamic parameters. This selected airfoil is used for further wing designing process.

2. METHODOLOGY

The objective is to achieve better lift from an airfoil which will be used for further applications. The research is aimed at selecting suitable airfoil for the construction of micro UAVs. All the parameters regarding airfoil are directly taken from the database i.e. only these parameters are compared with respect to airfoil. Starting with Reynolds number, various graphs from XFLR 5 are compared and at last ANSYS analysis is done. The procedure to use this software was studied online. The process for selection is shown in Fig. 1.

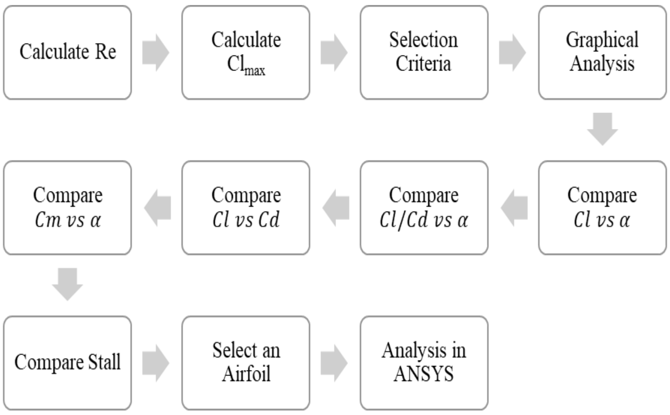


Fig. 1 Airfoil selection process

2.1 Case Study

Select an airfoil for the micro UAV wing lifting capacity of 2.5 kg with cruising and stalling speed 15 m/s and 10 m/s respectively.

2.2 Selection Process

The selection process is as per Fig 1. Parameters of the airfoil are compared i.e. (Cl vs α) graph, (Cd vs α) graph, (Cl/Cd vs α) graph, (Cm vs α) graph [3]. Finally, the best-chosen airfoil for the given input requirement is analyzed in ANSYS.

2.3 Calculate Reynolds Number (R_e)

R_e is used to determine whether the airflow is laminar or turbulent for analyzing airfoil, R_e depends on altitude, temperature, viscosity, velocity, density & mean aerodynamic chord [2].

1) Calculate Kinematic viscosity υ from International standards of atmosphere (ISA):

As per the conditions, Temperature (T) 15°C or 288 K & Altitude (A) 60 m, dynamic viscosity (μ) is 1.802×10⁻⁵ kg/ms & Density (ρ) 1.225 kg/m³ [4], Kinematic viscosity υ = μ/ρ =1.471×10⁻⁵ m²/s. The dynamic viscosity is decreases with increase in altitude & decreases with temperature.

2) Calculate R_e for cruising velocity (V_c) and stalling velocity (V_s) [5].

$$R_e = \frac{v \times mean\ aerodynamic\ chord\ (MAC)}{\vartheta}$$

Table 1 Calculation of Reynolds number

Sr. No.	Condition	v (m/s)	MAC (m)	ϑ (m²/s)	R_e
1	Cruising	15	0.21	1.471×10⁻⁵	2,14,140.04
2	Stalling	10			1,42,760.02

The airflow is turbulent as shown in Table 1, Low R_e can be achieved maximum lift coefficient [6]. Use R_e range 100000 to 250000 & consider N_{crit}=9.0 for the batch analysis in XFLR5 [7].

2.4 Calculate Cl_{max}

Firstly calculated wing area after iterations of calculations and then taken for Cl_{max} calculation as Weight (W) =24.52 N, Span (b) =1.30 m, chord (c) =0.21 m, aspect ratio (AR) =6.19, wing surface area (S) =0.2730 m², density (ρ) =1.225 kg/m³, V_c =15 m/s, V_s= 10 m/s.

(1) Calculate Aircraft Lift Coefficient for cruising (Cl_c) and stalling (Cl_s)[8].

$$Cl = \frac{2W}{\rho \times v^2 \times s}; Cl_c = \frac{2W}{\rho \times v_c^{\,2} \times s} = 0.65; Cl_s = \frac{2W}{\rho \times v_s^{\,2} \times s} = 1.46$$

(2) Calculate Wing Lift Coefficient for cruising (Cl_{c_w}) and stalling (Cl_{s_w}) [8]

$$Cl_w = \frac{Cl}{0.95}; Cl_{c_w} = \frac{Cl_c}{0.95} = 0.68; Cl_{s_w} = \frac{2W}{0.95} = 1.53$$

(3) Calculate Airfoil Lift Coefficient for cruising (Cl_{c_w}) and stalling (Cl_{s_w}) [8]

$$Cl_{\max} = \frac{Cl_w}{0.90}; Cl_{c_{\max}} = \frac{Cl_{c_w}}{0.90} = 0.75; Cl_{s_{\max}} = \frac{Cl_{s_{\max}}}{0.90} = 1.70$$

(4) Finally, calculate theoretical Airfoil maximum net Lift Coefficient (Cl_{max}) [8]

$$Cl_{\max} = Cl_{s_{\max}} - Cl_{c_{\max}} = 0.95$$

Finally, the airfoil parameters are shown in Table 2.

Table 2 Airfoil parameter with description

Sr. No.	Parameters	Description
1	Cl	Maximum
2	Cd	Minimum
3	Cl/Cd	Maximum
4	Cm	Lowest
5	Stall	Smooth nature
6	Thickness & Camber	Moderate

2.5 Airfoil Selection Criteria

The number of airfoils are available following suitable criteria to select the best airfoil for a UAV wing; for better lift capacity, airfoil selection keep Cl greater than theoretical Cl_{max}, for that add 0.50 to Cl i.e. 0.50+0.95=1.45, Hence Cl should be greater than 1.45. Select airfoil with moderate camber & thickness i.e.; camber should be greater than 5% of chord, the thickness should be greater than 10%, & (Cl/Cd) max should be greater than 60. Also, it should be structurally reinforced & manufactured. The α was selected as 4-5° for fixing on the fuselage, as it provides safety due to stalling i.e. from 10° most of the airfoil starts their stalling region.

2.6 Graphical Analysis

Now we will move forward towards graph analysis in XFLR 5 and find out maximum or minimum point and smoothness of curve as per our requirement. There are mainly four graphs as explained below;

2.6.1 Cl Vs α:

In Fig. 2 α is the stall angle where lift efficiency will no longer increase with α. Stall angle is an important factor for safety.

Airfoil has a higher stall angle, keeps aircraft flight safe. The stall angle ranges from 12° to 16°. Zero α at which Cl is zero. The design aim is to have a higher α greater than zero.[8]. Higher Cl_{max} results in decreases stall speed so for safe flight higher Cl_{max} is required. The ideal coefficient of lift (Cl_i) is the Cl where Cd does not vary significantly with minor deviations of α. Cl_i is corresponding to lower drag. Here it is necessary for low flight cost. The lift coefficient at zero angle of attack (Cl_0) is the Cl at α is zero. At High, Cl_0 is better to design as it can produce a positive lift at a zero α [8]. The nature of the Cl curve at and beyond stall angle should be smooth. Airfoil has a gentle drop in the lift after stall leads to safer stall from which pilot can recover.

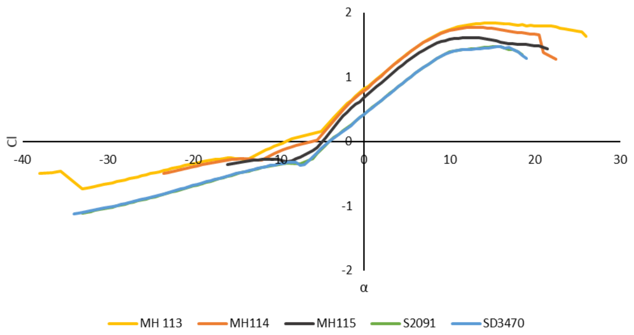


Fig. 2 Graph Cl vs α

2.6.2 Cm Vs α:

The slope of Fig. 3 is usually negative. The negative slope is desired, it stabilizes flight even if α is disturbed. The magnitude of Cm is constant at the range of -0.02 to -0.05 [8]. The value of Cm is nullified by the tail component. The Cm should be close to zero as far as possible to have equilibrium in flight [8].

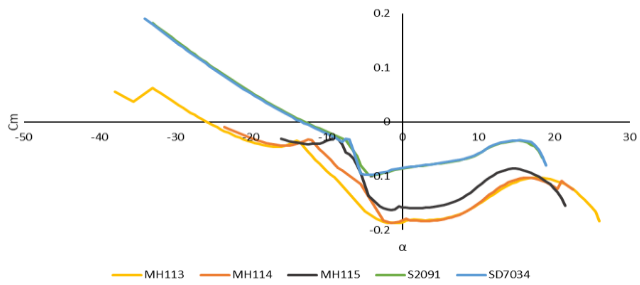


Fig. 3 Graph Cm vs α

2.6.3 Cl vs Cd:

The lowest point in Fig. 4 shows the minimum drag coefficient Cd_{min} and the corresponding lift coefficient is Cl_{min}. The minimum value of Cd_{min} is required and it ranges from 0.003 to 0.006 [8]. The bucket shape of the lower region of the graph is a unique feature [8]. The Cd_{min} will not vary for a limited range of Cl. It indicates that a pilot can stay at the lowest drag region however changing α [9]. Hence the pilot can decrease α without increasing drag.

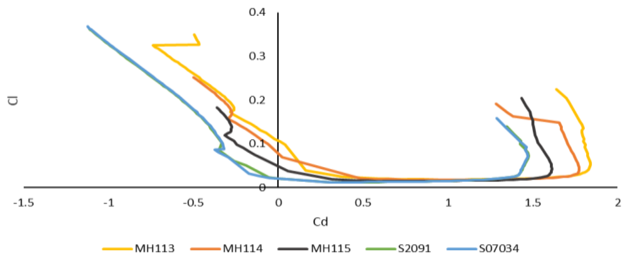


Fig. 4 Graph Cl vs Cd

2.6.4 Cl/Cd Vs α:

Fig. 5 has covered almost every parameter for comparison. At an extreme point, whether the value of the lift to drag ratio is maximum, α at this point becomes optimum. The slope of Fig. 5 should be such that to have a higher α at the stall region.

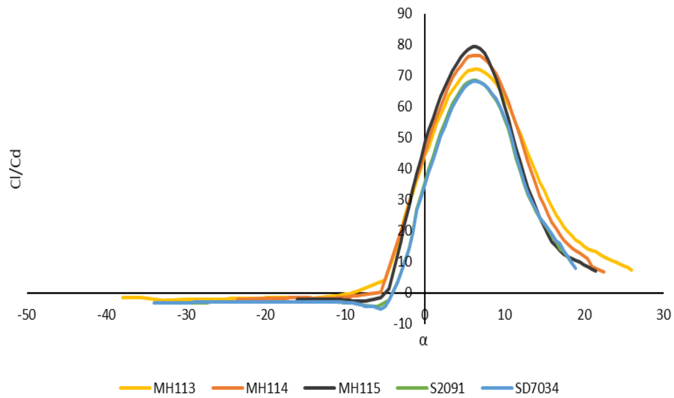


Fig.5 Graph Cl/Cd vs α

3. RESULT

Considering all constraints for the application of plane finally an optimized airfoils were selected for the UAV wing after calculations and parameter curves.

Table 3 Observation of airfoils

Sr. No.	Airfoils	Cl_{max}	$(Cl/Cd)_{max}$	Alpha stalling	Stalling
1	Martin Hepperle 113 (MH113)	1.820	72.149	7.123	Docile
2	Martin Hepperle 114 (MH114)	1.727	76.435	5.758	Sharp
3	Martin Hepperle 115 (MH115)	1.588	79.484	6.037	Moderate
4	Selig 2091 (S2091)	1.406	68.588	6.156	Sharp
5	Selig-Donovan (SD7034)	1.381	68.347	6.321	Moderate

From Table 3, we have selected a MH113 for the wing. This airfoil has a smooth docile curve at stalling with maximum Cl_{max}. It has maximum $(Cl/Cd)_{max}$, so it produces minimum drag with higher stalling α. Airfoil Type: Cambered Airfoil, Selected Airfoil Name: MH113, Thickness: 14.67%, Camber: 6.90%.

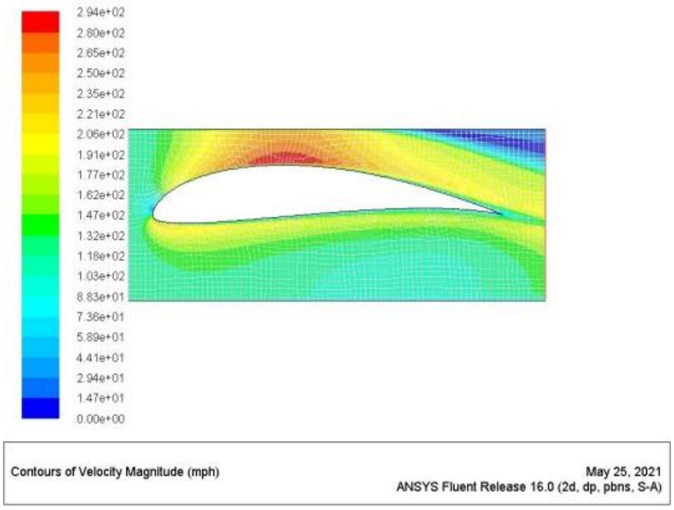


Fig. 6 Velocity distribution over airfoil profile

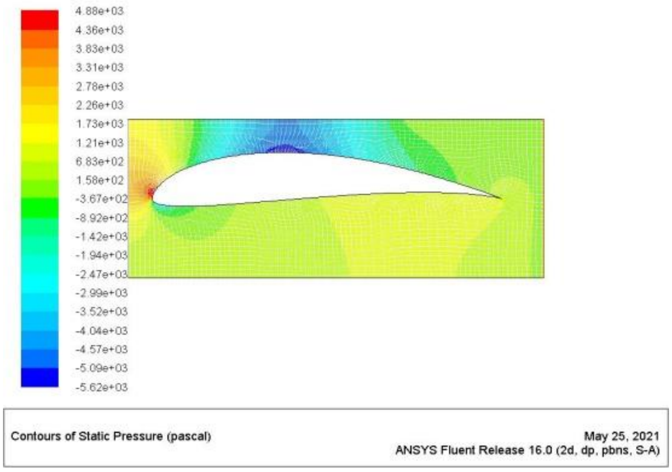
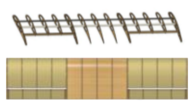


Fig.7 Pressure distribution over airfoil profile

The selected airfoil MH113 is analyzed in ANSYS at α= 0° and pressure and velocity distribution is generated as shown in Fig. 6-7. Here flow of air at upper surface and pressure distribution along the chord of the airfoil is smooth and changes with the angle of attack.

We reached to stage where selected a suitable airfoil, actually fabricated this model which worked efficiently. We have discussed only the airfoil selection procedure which is part of the work as can be seen in Table 4 but the complete design and fabrication of the RC plane is another major work that is not part of this research. The material used for the fabrication of airfoil is balsa wood glued with adhesives and a power source as an electronic battery.

Table 4 Airfoil placement in UAV wing

Sr. No.	Parts	CAD Model	Actual Model
1	Airfoil		
2	Wing		
3	RC Plane		

4. DISCUSSION

Reynolds number plays an important role in analysis so calculated for both conditions i.e. cruising and stalling & Cl vs alpha is the most important graph for the determination of Cl as stalling can be seen. Pressure distribution generated from ANSYS proves that MH 113 has a good lift as predicted from the graph. This selection is a part of our project so practically RC plane flight is efficient with respect to lift.

5. CONCLUSION

The proper method for selecting symmetric & unsymmetric airfoil on the basis of selection criteria and analyze it provides us a better idea as to which airfoil is appropriate for the defined parameters before designing a micro UAV wing. The selection process is quite flexible and a suitable airfoil can be obtained for different applications with input as appropriate constraints, these all we have elaborated with examples in this work. The smooth graphical nature of aerodynamic characteristics gives a better overall performance in a lift. The major part is performed by graphs from XFLR5 and analyzed from ANSYS.

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