Research Journal of Engineering Sciences ISSN 2278 – 9472
Vol. **9(1),** 19-28, January **(2020)** *Res. J. Engineering Sci.*

Design of fixed wing RC aircraft

Khanal Aakarshan
Department of Mechanical Engineering, Institute of Engineering, Purwanchal Campus, Dharan, Nepal
aakusanu@gmail.com

Available online at: www.isca.in, www.isca.me
Received 29th August 2019, revised 13th November 2019, accepted 15th January 2020

Abstract

*This paper can be a guide for designers interested in aeronautics for learning basic design ideas. This paper contains design of high wing, conventional tail, tricycle gear RC model with tractor configuration. This paper mainly concerns on design of RC aircraft which can be used for specialized purposes like gathering weather information, spy planes and experiments. Nose mounted motor with propeller of 8*4.5 inch is used as propulsion unit. Dihedral wing configuration is set up with a dihedral angle of 3° using Clark Y airfoil which has wing span of 96cm, chord length of 19.2cm and the aspect ratio of 5. Designed fuselage has the length of 76cm which holds the electronics component like battery and the receiver. 3D model of the design is developed in Solid Works 2016 followed by simulation of dihedral wing with air velocity of 10m/s at 5° AOA in ANSYS CFX. Simulation result showed that wing produces lift of 6.7248 N. XLFR5 was used for analysis of airfoil and model aircraft. Propulsion system analysis is carried out in MotoCalc to select best power combination for desired performance level.*

Keywords: Dihedral, XFLR5, ANSYS CFX, MotoCalc.

Introduction

While, field of design of RC aircraft is relatively new in context of Nepal, study materials regarding the design and fabrication of RC aircraft exist in the literature. To get the main objective, this research focuses on sub objectives like development of 3D cad model, analyzing the performance and stability of designed aircraft and performance simulation of designed aircraft wing to determine lift and drag. Aileron, elevator and rudder are control surfaces which assists aircraft in maneuvering. Radio controlled aircrafts mostly have straight edged wings while delta wing can also be found in special cases i.e. when large speed is the requirement for aircraft[1]. Radio controlled aircrafts are designed to study the feasibility of the configuration and various performance characteristics. CFD analysis on the other hand provides a mean to study the performance of aircraft in real flying conditions through simulation[2]. There are lots of significance of RC aircraft in country like Nepal. Scientific and government organization can utilize radio-controlled aircraft for collecting weather information and laboratory purpose for experiments. Likewise, military organizations can use RC aircrafts as drones or spy planes.

Methodology of Design

There are mainly three phases in methodology of design[3-5]. i. Conceptual Design, ii. Preliminary Design, iii. Detailed Design.

Modeling and Analysis: AutoCAD and Solid works can be used for developing 3D model of the freeze configuration followed by performance analysis which can be done in XFLR5, which is an open source software mostly used by engineers for designing RC airplane. XFLR5 is free source software for analysis of airplane and airfoils[6]. During development phase of model, based on equations and formulas provided in literature documents, calculation of dimension for various parts of the aircraft are done. Solid works 2016 was chosen for modeling of Airframe.

Design Research: Conceptual Design: Sizing and Takeoff: Since, the material used for RC aircraft is low density material, the length of wing should be estimated such that it prevents from bending. Taking into consideration the availability of open space, the rolling distance must be maximum of 20m. Thus, maximum size of the airplane is estimated to be about 700gm.

Propulsion: Internal combustion engine and Nitro engine are not feasible for RC model with light density material like Styrofoam due to working temperature. Thus, the best option is using electric motor with propeller as propulsion unit.

Configuration Selection: Seven configurations were selected for study from a bunch of available configurations. First configuration is conventional tail with high wing. The model has one motor which is nose mounted i.e. tractor configuration. Second configuration is also conventional tail with high wing. The model has one motor which is pusher configuration. Third configuration is also conventional tail with high wing. The model has two motors mounted at wings. Fourth configuration is conventional tail with mid wing. The model has one motor which is nose mounted i.e. tractor configuration. Fifth configuration is also conventional tail type with mid wing. The model has one motor which is pusher configuration. Sixth configuration is also conventional tail type with mid wing. The model has two motors mounted at wings. Seventh configuration is flying wing. The main concept of this configuration is to increase wing area. This, configuration also provided platform to compare conventional tail configuration with tailless configuration.

Conclusion: Good stability and control were major benefit for selecting conventional tail as best option. Moreover, selecting the position of the wing, high wing design is appropriate because it gives huge space in fuselage. Low or mid wing configuration would have metallic or wooden rod used for support passing through the fuselage thus obstructing the battery placement. As a result, battery would move backward thus affecting the position of center of gravity. Single motor was only option due to weight factor though single motor have lower performance than two motor design. The motor position was nose mounted. Thus, the conceptual design freezes conventional tail, single motor and high wing design.

Preliminary Design: In preliminary analysis, conventional tail high wing configuration and its components were examined. Propulsion unit and aerodynamics study were done in details.

Aerodynamics: Lift produced is proportional to wing area and having larger wingspan produces greater lift[7-9]. Takeoff distance is affected by different variables and reference were taken from published thesis report[10]. Chord length, TR, wing position from ground are some of the variables. Thus, factors we must give emphasis on is weight of airplane and chord. It is necessary to select appropriate chord length else too small chord will cause airplane to takeoff from distance longer than the feasibility of airfield.

Moreover, weight is major factor in affecting takeoff distance and the design should be with lowest possible weight. Since ground effect and taper ratio has negligible effect in takeoff distance[10], thus selection of high or mid wing is highly influenced by ease of fabrication rather than the performance. Thus, high wing is better option.

Center of gravity based on electronic components layout: Estimate of placement of electronic components gives the location of center of gravity which can directly fluctuate the stability. So, COG must be analyzed in detail. At first, the position of center of gravity should be approximated and placement of electronic components must be done as per the approximation. XFLR5 is used to determine the position of center of gravity during preliminary design.

Preliminary propulsion system analysis: Preliminary propulsion system analysis requires listed consideration. The main reason behind preliminary analysis of propulsion system is to determine which type of motor will require least battery weight as well as overall propulsion system weight. Number of Motors: Large size single motor is selected due to ease of installation compared to numerous small sized motors. Motor type: Out runner brushless motors is the choice due to low maintenance, high efficiency and higher torque. Propeller type: Narrower blade props is preferred due to better climbing performance and better efficiency at top speed and maximum thrust compared to wider blade props. Number of propeller blades: Though a propeller with higher number of blades will perform better, two bladed propellers is considered based on availability in market. Motor mounting location: In general, there are two configurations for mounting motor: pusher and tractor configuration.

Motor is installed with backward thrust in tractor design whereas in pusher design it is vice versa. Tractor design is preferred due to easy installment. Driving mechanism: Propellers can be driven by motor either directly or by gear drives. Gear drive is not preferred due to complexity and more frequent maintenance. Thus, it is better to use direct drive with ESC installed to control the rotation speed of the motor.

Conclusion: Preliminary design concluded with decision to use high wing with large chord length and aspect ratio. Taper ratio of wing was decided to be zero. Out runner brushless motor was final option with two bladed maximum thrust type propellers rotated by direct drive mechanism.

Detailed Design

Wing: Airfoil Choice for Wing: Main parameter that decides the choice of airfoil are camber and thickness. Initially some favorable and widely used airfoils were screened from hundreds of airfoils available these days based on literature and successful flight of the particular model mentioned in the literature. An airfoil profile is generally highly cambered, moderately cambered, or symmetrical. The down pitch moment of an airfoil increases with the increasing camber. Where a high camber provides greater CL and CM, and no camber/symmetry generates zero pitching moment; a moderate camber is the most preferred, because it gives enough lift with not much drag and has a gentle stall pattern in contrast to a sharp stall pattern for others. Following are the list of symmetrical and unsymmetrical airfoils screened based on literature review. NACA 0018, NACA 0012, Clark Y, NACA 1412, NACA 2412, NACA 3412.

Analysis and Result: After finalizing the camber range, some of the most popular moderately cambered airfoils were tested and compared using the analysis on XFLR5. Figure shows coefficient of lift for different airfoil at different angle of attack. As observed from the polar curves, Clark Y displayed an extraordinary behavior in comparison to the other three. Considering the obtained results, Clark Y airfoil has value of C_L =1.01 at 6° AOA which is higher compared to other airfoils. Moreover, at this AOA value of C_D is 0.1943 which is the least among enlisted airfoils. Thus, from above results Clark Y showed better performance characteristics and was the final choice of airfoil. The stalling AOA is very high and the coefficient of lift is exceedingly good though it has a pitch down moment, its other benefits overcome that factor.

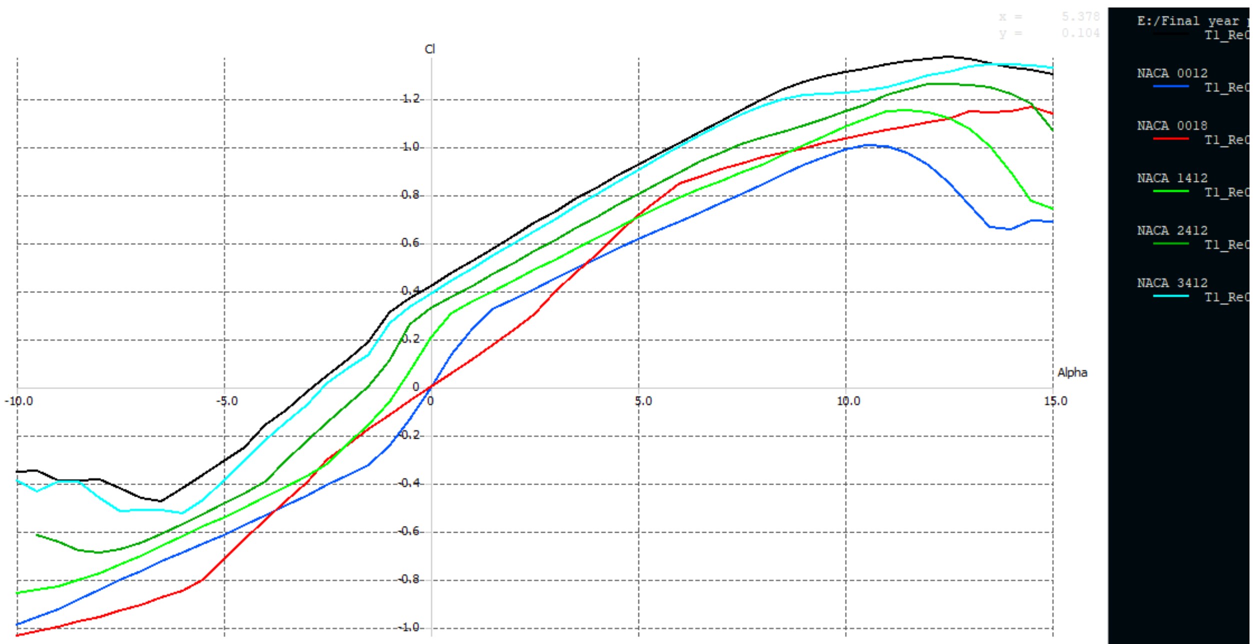

Figure-1: CL vs Alpha plot for various airfoils (XFLR5)[6].

Wing Loading: Flight behavior and maneuverability was studied for a range of wing loadings and following data were obtained[11,12]. Since the model designed is a low speed trainer, a maximum wing loading of 0.488gm/cm^2 is acceptable.

Aspect Ratio: Low and high AR both have their own benefits and limitations. If the wingspan increases, it becomes more difficult to support a wing as the bending moment increases about the attachment point. High aspect ratio gives high lift but develops more friction drag due to increased wing span[13]. A low AR wing has high induced drag in comparison with a high AR wing. Compromising on the merits and demerits of both the kinds, a moderate aspect ratio of 5 was found to be most convincing.

Angle of Incidence Calculations

$$\alpha = \alpha_0 + \frac{18.24\, C_L}{AR} \qquad (1)$$

Where, α = Total angle of attack, α_0 = Section/airfoil AOA relative to zero lift AOA, C_L = required lift coefficient and AR = Wing Aspect Ratio from C_L vs Alpha curve of Clark Y airfoil, we can find that zero lift occurs at -2.8° AOA. For a level flight at 10m/s coefficient of lift has a value of 0.608 and total angle of attack=8.01 and angle of incidence= 8.01-2.8=5.21. Thus, we select 5° angle of incidence for wing.

Wing Planform: There are mainly three planforms commonly used in aircraft: Elliptical, rectangular and tapered. Elliptical wing is considered ideal one because it has low induced drag and stalls evenly along the span. Fabrication of elliptical wing is tedious. Another type is rectangular wing, which is very easy to fabricate and also have sensitive control surface. A tapered wing offers a benefit of lower tip losses in comparison with a rectangular wing, due to low Reynolds number at the tips. But tip stalls occur in a tapered wing more readily.

Lateral Stability: Dihedral is an upward slope given to the wing planform along the wing span symmetrically about the wing root. A dihedral angle is known for its usage in roll or lateral stability of a plane. So, if the aircraft is disturbed or banked in one direction, say to the starboard side, then an additional sideslip motion of air occurs from tip to root on the lower wing. Here, the dihedral angle provides an AOA to the sideslip, thus net lift generated to starboard being more than that on the port side. A restoring moment thus acts to stabilize the aircraft and brings it back to an upright position. In the absence of a dihedral, it would slip to the side and go down if not corrected.

In a high wing setting, the CG is below the wing. This allows the center of pressure to remain above the CG and so, the lift force and weight form a restoring couple in case of a disturbance, providing the inherent stability to the wing. A large dihedral angle reduces net lift, increases the drag and the lateral forces leading to a side-to-side roll or commonly known as 'Dutch Roll'. So, a small dihedral angle3° was chosen for model, owing to its high wing configuration.

Design of Wing: Initial assumption needs to be made before the initiation of design process. Thus, the weight of aircraft is considered to be 700gm, wingspan is 96cm and aspect ratio (AR) is 5. From the range of aspect ratio for different types of aircraft, the value of 5 was selected for the model[13,14].

Step 1: To determine the plan form area (S_{wing}) [12,14,15]. $S_{wing} = \frac{b^2}{AR}$, $AR = \frac{b^2}{S_{wing}}$ $S_{wing}=96^2/5=1843.2cm^2$	**Step 2**: Calculation of wing loading[12,14,15]. Generally, wing loading is considered to be below $0.6gm/cm^2$. Wing Loading $= \frac{\text{Mass of aircraft}}{S_{wing}}$=700/1843.2 =$0.3797gm/cm^2$
Step 3: Taper ratio (TR) = 1 (Straight wing will be used).	**Step 4**: $C_{root} = \frac{2^*S_{wing}}{(b^*(1+TR))}$ [12,14,15] = 2*1843.2/ (96*2) =19.2cm
Step 5: $C_{tip} = (TR^*C_{root})$[12,14,15]= (1*19.2) =19.2cm Check: Wingspan should be 5 to 6 times of Chord. 19.2*5=96 cm (Correct)	**Step 6**: To find the value of $C_{root'}$. $C_{root'}$ represents chord at middle of the wingspan. For straight wing, the value of $C_{root}=C_{tip}$. Therefore, $C_{root'}$=19.2cm
Step 7: MAC (C’’). MAC is defined as chord where total lift generally acts[12,14,15]. $C'' = \frac{2C_{root}(1+TR^2+TR)}{(3(1+TR))}$ So, C’’=19.2cm	**Step 8**: Aerodynamic centre (X_{ac}). It is assumed to be point along the wing width where entire lift force act. Aerodynamic center always lies in Mean aerodynamic chord[12,14,15]. $X_{ac} = (C_{root} - C') + \frac{C'}{4}$= (19.2-19.2) +19.2/4= 0+4.8 Therefore, X=4.8cm
Step 9: Design of Airfoil: Equating aircraft weight to lift $W = L$, $1/2^*V^{2*}\rho^*C_L{}^*S_{Wing} = g^*M$ Density of air (ρ) = $1.225kg/m^3=1.225*10^{-3}g/cm^3$, acceleration due to gravity (g) = $9.81m/s^2 = 9.81*10^2cm/s^2$, cruise speed (V) = 10m/s = $10*10^2$cm/s, S_{wing}= $1843.2cm^2$, M= 700g. Therefore, C_{L3D}=0.608.	

Computational Fluid Dynamics (CFD) analysis of fluid flow over wing: Simulation of the designed model is done to visualize lift, drag and streamlines. Due to the complexity of equation governing flow over wing, it’s hard to find solution straightly. Thus, flow simulation is very important. The modelling and meshing of wing were done using ANSYS CFX package. Emphasis in simulation is given to velocity distribution, pressure distribution, lift and drag. Dihedral wing was modelled and meshed as fluid domain. Tetrahedron element shape was used for the meshing (discretization) of the model.

Boundary Condition: The simulation has been done to investigate velocity and pressure distribution across the wing chord and to determine value of lift and drag. The main assumption includes flow in steady manner and zero slip condition. The fluid was air (density = 1.225 kg/m^3). Velocity of air flowing across wing was taken as 10m/s and pressure at trailing edge as atmospheric pressure. The velocity streamline was generated by ANSYS 15.0. Result shows that velocity is maximum at maximum thickness point. Moreover, streamline animation was viewed in same software and it is found that flow across wing surface as required i.e. flow separation doesn’t occur. Moreover, lift developed in the wing is 6.7248 N and drag along the flow direction is 0.543679 N.

Horizontal and Vertical Stabilizer: Horizontal stabilizer maintains pitch stability. Parameters which governs pitch stability are the tail moment arm, tail area, tail mounting location, elevator area and its throw. Following decisions were taken regarding these factors.

Airfoil Selections: In this research, a certain innovation is made in context of airfoil for horizontal and vertical stabilizer. Thus, a decision to test for a unique type of airfoil was made, that was designed. Initial model was designed using NACA 0018 airfoil for H.S and V.S. Rudder and elevator were found to be less effective due to thickness of airfoil. Moreover, H.S and V.S had no contribution to lift due to smaller span. Designed airfoil was nothing but a nearly flat airfoil with 0.6 % thickness at 0.1 %. Characteristics of airfoil were analyzed with Reynolds number 131500 using X Foil Direct Analysis in XLFR5. The results are shown in figure below. From result one can find that designed airfoil produces zero lift after 7° angle of attack. It means the horizontal and vertical stabilizer stall after 7° angle of attack. But designed airfoil shows excellent performance in terms of drag. Though the airfoil has zero lift at 7° but has C_d=0.07 at this angle of attack.

Thus, the horizontal and vertical stabilizer although stalled doesn’t contribute to drag. Moreover, designed airfoil is easier to fabricate from Depron and even though having thickness of 6 mm can withstand the air pressure easily. Thus, designed airfoil is selected for H.S and V.S.

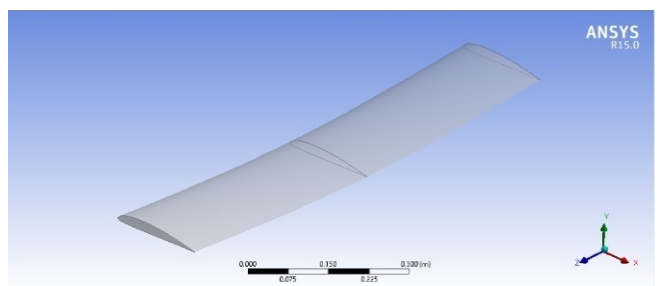

Figure-2: Modelling of wing in ANSYS Design Modeler[16].

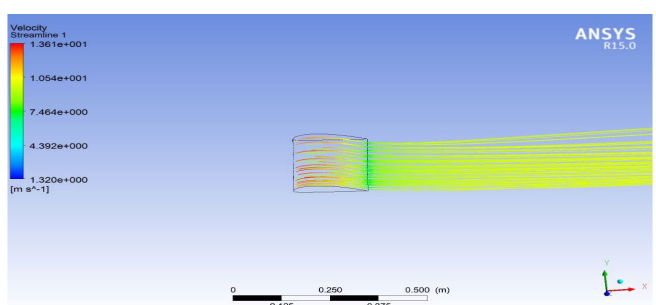

Figure-3: Velocity streamline for wing (ANSYS)[16].

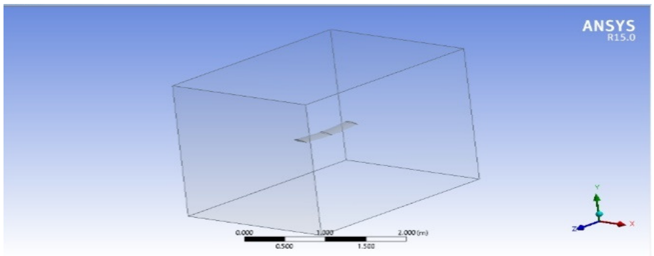

Figure-4: Enclosure of 1m*1m*1m for simulation (ANSYS)[16].

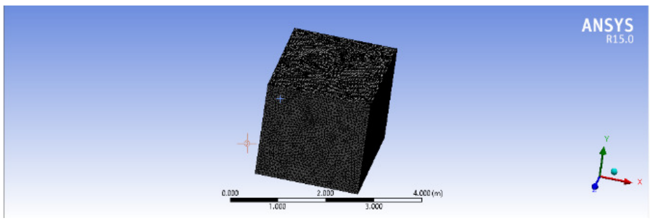

Figure-5: Meshing of wing model (ANSYS)[16].

Tail Position: Wake generated at the wings shouldn't be encountered by the tail else the efficiency of tail decreases. Thus, tail is placed at a certain clearance height from the wing. The inverted T-tail configuration was only opted for, owing to more structural strength requirements for the vertical stabilizer and more reinforcement that should be done to the horizontal stabilizer when using a T-tail.

Neutral Point and Static Margin: Neutral point is referred to as the point where the net effective aerodynamic forces act after summing up the lift and drag on wing and tail. The neutral point location depends on the tail moment arm from the CG and is preferably located at 35% chord length from the wing leading edge. Static margin is distance between CG and neutral point. A positive static margin gives the aircraft inherent longitudinal stability. For example, if there is a pitch up disturbance on the plane, the net lift acts behind the CG to produce restoring pitch down moment. Thus, a greater tail moment arm from the CG was selected.

Table-1: Design of Horizontal Stabilizer.

Step 1: Horizontal Stabilizer area (S_t) is chosen as 20% of the area of wing considering stability as factor[8,15]. S_t = 20% of S_{wing}=0.2*1843.2=368.64cm^2	
Step 2: AR is selected lower than wing to delay stall compared to wing[8,15]. Tail aspect ratio (AR_t) should be between 3-5; So, AR=3.	**Step 3**: Tail span[8,15] $b_t = (AR_t{}^*S_t)^{1/2}$, [$AR_t$=3 considered] Therefore, $b_t = (3*368.64)^{1/2}$ b_t =33.25 cm
Step 4: Tail taper ratio (TR_t)=0.66 [considered]	**Step 5**: Root chord of tail Croot[8,15] $C_{root,t=}\frac{2^*S_t}{(b_t(1+TR))}$ $C_{root,t}$=2*368.64/(33.25(1+0.66)) $C_{root\ t}$=13.3617cm
Step 6: Tail tip chord [8,15] $C_{tip} = TR_t{}^*C_{root,t}$ = (0.66*13.3617) Hence C_{tip}=8.8187cm	**Step 7**: Setting of tail is considered three degree lower than AOA. i_t=5-3=2
Step 8: Elevator area is always 20% of stabilizer area =0.2*368.64=73.73 cm^2. Elevator width=Elevator area/Stab length= 2.22cm.	

Table-2: Design of Vertical Stabilizer (VS).

Step 1: Area of fin = 30-35% stabilizer area Area of fin =33*368.64/100 =121.65cm^2 Fin height=19.11cm Average fin width= 6.37cm	**Step 2**: To calculate Rudder area Rudder area is 1/2 to 1/3 of fin area=0.333*121.65=40.1445 cm^2 Average Rudder width=2.1cm.

Design of Winglet: Winglet height is given by following equation[17]. Winglet height=Winglet length=Chord length of wing=19.2cm.

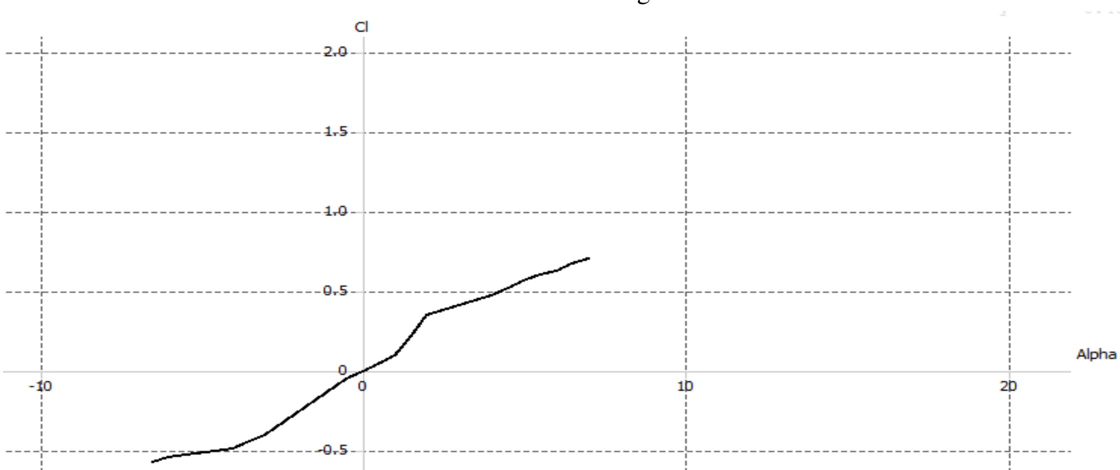

Figure-6: C_L vs Alpha for designed airfoil (XFLR5)[9].

Servo Sizing: For sizing servo for aileron, elevator and rudder, drag force analysis is the easiest method18. Drag calculation was done in Solid Works Flow simulation, where the analysis was done considering each component as separate contributor to drag i.e. while analyzing aileron wing wasn't considered in simulation. In other word, drag analysis of aileron was done without taking wing into consideration. Doing so we can get maximum drag the components can give during flight. For drag analysis the velocity of air was taken same as for the aircraft i.e. 10m/s.

Considering 20-degree elevator throw18. Net drag force on elevator was calculated to be 0.2545 N and assuming this force to be acting at the trailing edge of the elevator. So, net torque required from the servo motor to counter 0.2545N at 2.22cm from the elevator hinge (i.e. at the trailing edge) is = 0.2545/9.81 * 2.22 * sin 20 = 0.01969Kgf-cm.

Though the torque requirement is not much, the elevator when working with vertical stabilizer encounters greater drag force.

Considering 10-degree aileron throw[18]. Net drag force on aileron was calculated to be 0.102936 N and assuming this force to be acting at the trailing edge of the aileron. So, net torque required from the servo motor to counter 0.102936N at 4.8cm from the aileron hinge (i.e. at the trailing edge) = 0.107936/9.81 * 4.8 * sin 20 = 0.018 Kgf-cm

Assuming this force to be acting on the aileron at the trailing edge (extreme end from the hinge), maximum torque required from the servo motor, to keep the flap in equilibrium, was obtained.

Considering 20-degree rudder throw[18]. Net drag force on aileron was calculated to be 0.125358N and assuming this force to be acting at the trailing edge of the rudder. So, net torque required from the servo motor to counter 0.125358N at 2.1 cm from the rudder hinge (i.e. at the trailing edge) is = 0.125358/9.81 * 2.1 * sin 20 = 0.00917Kgf-cm

Thus, servo motor with torque greater than 0.1969Kgf-cm is acceptable.

Design of Fuselage: Length of fuselage is taken to be about 80% of wingspan. Length of fuselage ahead of wing leading edge=1 to 1.5* chord=19.2cm and Length of fuselage behind trailing edge=2 to 3 * chord= 38.4cm

Other aspects: It is good to use dihedral angle of 3 degrees for better stability of the aircraft. Moreover, in tricycle landing gear used, the main gear should be slightly aft of the balance point in order to get easier takeoff. To increase the control surface throw, the push rod is moved to the hole on the control horn that is closer to the control surface and moved push rod to the further out hole on the servo arm.

Mass distribution and CG Balancing

Styrofoam available in market has density ranging from 12 kg/m^3 to $14kg/m^3$. Density of $13kg/m^3$ is used for convenience. Likewise, Depron found locally has density ranging from 48 kg/m^3 to $50kg/m^3$. Thus, a density of $49kg/m^3$ is used. Styrofoam is used for fabricating the main wing and fuselage whereas Depron is used to build control surface and empennage. Though, Depron has better strength than Styrofoam, Styrofoam is preferred in fuselage and main wing due to ease of fabrication and reduced weight.

The power plant used for the model is A2212/10T, which has a weight of 62gm. The servo used weighs 10gm each. For accounting additional weight 20gm is added to each section of plane. The additional weight can be due to glue and cartoon tape used in that particular section. Different weights for various components like motor, battery, servo motors, etc. were added in the fuselage at appropriate locations, so as to locate the CG near the aerodynamic center.

Mass of different components

Wing =44gm, Battery=180gm, Fuselage=46gm, Front landing gear=32gm, Servo=10gm, Motor=68gm, Prop=10gm, ESC=24 gm, Main landing gear=76gm, receiver=16gm, wire=80gm, HS=10gm, VS=4gm and Glue=100gm.

At this point mass for different component is finalized and aircraft overall weight was decided as 700gm approximate and center of gravity is located at 4.83cm from leading edge of the wing along chord from XFLR5.

Aerodynamic Analysis: Fuselage has no such contribution in performance of RC airplane except storing payload and increasing drag. Though some designers tend to build lifting fuselage but using low density material as well as making fuselage to contribute in lift can hamper the structural strength of the airplane. So, in this particular aerodynamic analysis section fuselage has been omitted from analysis. A detailed analysis of wing and tail section is done in XFLR[5]. It is not a necessity to analyze the whole structure of the airplane since it can be cumbersome and not so much effective for performance analysis.

Aircraft analysis using Horseshoe Vortex (VLM 1): VLM 1 analysis was carried with airspeed of 10m/s, air density of $1.225kg/m^3$, temperature as 25°C and altitude of 200m. Analysis was carried out for range of angle of attack starting from -5 degree to 10 degrees. XFLR5 analyzed the aircraft and provided the result at each angle of attack. In the analysis process fuselage hasn't been considered due to limitations of software.

Table-3: Coefficient of drag and lift at different AOA by VLM 1 and VLM2 (XFLR5)[6].

VLM1			VLM2		
AOA	Coefficient of Lift	C_D	AOA	Coefficient of Lift	Induced Drag coefficient
-5	-0.174	0.003	-5	-0.175	0.003
-4	-0.096	0.002	-4	-0.097	0.001
-3	-0.018	0.001	-3	-0.020	0.001
-2	0.060	0.001	-2	0.058	0.001
-1	0.138	0.002	-1	0.136	0.002
0	0.216	0.003	0	0.213	0.003
1	0.294	0.006	1	0.291	0.006
2	0.371	0.009	2	0.368	0.009
3	0.449	0.013	3	0.445	0.012
4	0.526	0.017	4	0.521	0.017
5	0.602	0.023	5	0.597	0.022
6	0.678	0.029	6	0.673	0.028
7	0.754	0.035	7	0.749	0.035
8	0.829	0.043	8	0.823	0.042
9	0.904	0.051	9	0.897	0.051
10	0.978	0.060	10	0.971	0.059

Analysis showed that, aircraft doesn't stall upto 10-degree angle of attack, thus the previous calculation of stall AOA= 11° is correct. Since $C_{m\alpha}<0$ and $C_{mo}<0$, for the aircraft, thus the aircraft cannot be trimmed.

Aircraft analysis using Ring Vortex (VLM 2): VLM 2 analysis was carried with airspeed of 10 m/s, air density of 1.225kg/m^3, temperature as 25°C and altitude of 200m. Analysis was carried out for range of angle of attack starting from -5 degree to 10 degrees. XFLR5 analyzed the aircraft and provided the result at each AOA. Drag and lift coefficient at each AOA are shown in the table.

Analysis showed that, aircraft doesn't stall up to 10-degree angle of attack, thus the previous calculation of stall AOA= 11 is correct.

Further analysis of propulsion

The major parameters taken into consideration while selecting propulsion unit is mass of motor, battery weight and so on. For making the selection of propulsion unit easier, selection of motor for comparison from large number of motor available in market were than classified under three categories: Small, Medium and Large. Large motor could propel the airplane itself. Medium motor needed one pair to propel the aircraft and small motor required more than one pair to propel. Since, the model was a nose mounted propulsion type, our decision is around the large motors only.

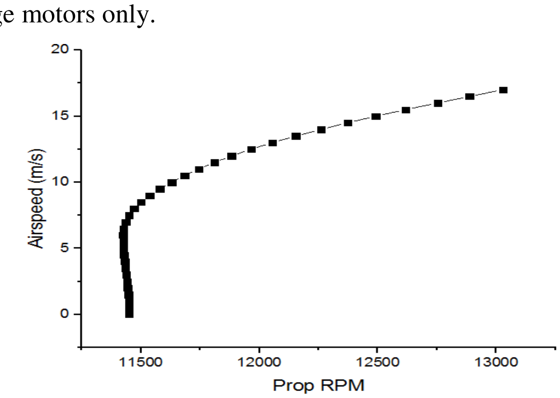

Figure-7: Airspeed vs Prop RPM for BP A2212/10T (MotoCalc)[19].

Thus, the list ended up in two names: Suppo A2212/10T and BP A2212/10T. MotoCalc software was used for further analysis of propulsion unit. After complete analysis BP A2212/10T was found to be superior in performance compared to other one. Thus, it was decided to select BP A2212/10T. Analysis showed that power combination using BP A2212/10T can produce lift sufficient for level flight at 10m/s. From the plot, it shows that BP A2212/10T with selected combination can provide airspeed of 10 m/s at 11500 RPM.

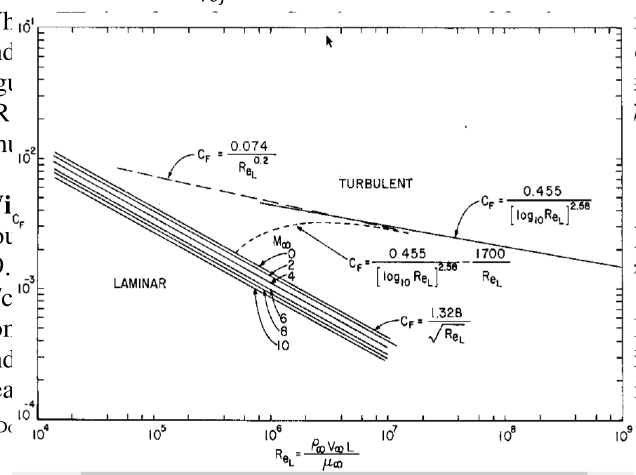

Figure-8: Coefficient of friction at various Reynolds number[20].

Stability test: Stability test on XFLR5 with designed model showed that the aircraft has both longitudinal and lateral stability. Longitudinal stability test was carried under two circumstance: Pitch up and Pitch down gust and aircraft should good stability by pitching back to original state of level flight. Likewise, Lateral stability test was carried under two circumstance: Roll left and right and aircraft should good stability by rolling back to original state of level flight.

Model drag estimation: We can find polar of drag for particular airplane by the expression given below[20].

$$C_D = C_{Do} + \frac{C_L^2}{\pi\, AR\, e} + k(C_L - C_{LO})^2 \qquad (2)$$

The equation is slightly different from wing drags polar equation. The first term C_{Do} collectively represents skin friction and pressure drag from different components. The C_{Do} is called skin friction and the data from figure below is used in its calculation. For Re <5*10^5 there is laminar boundary layer. By Re = 5*10^6, turbulent boundary layer exist.

Drag estimation: Fuselage length = 76cm, Fuselage width = 7cm, Wing aspect ratio = 5, Area of wing = S_{wing} = 1843.2cm^2, Wing span = 96cm, Landing gear: Tricycle landing gear (two main landing wheel and a nose wheel), Weight during takeoff = 700 gm, Airspeed=10m/s, Fuselage planform area=515cm^2, Fuselage Wetted area= 2060cm^2, Fuselage reference length=96 cm, Wing planform area=1843.2cm^2, Wing wetted area = 3686.4cm^2, Wing reference length = 19.2(MAC), HS planform area=368.64cm^2, HS wetted area = 737.28cm^2 and HS reference length = 11.09(MAC).

Fuselage: Reynolds Number (Re) =ρVL/μ= 6.573*10^7, so the boundary layer is turbulent. Where, Characteristic length (L) = 96 cm.

$$Fuselage\ C_{Do} = \frac{FF_F C_F S_{wet}}{S_{ref}} \qquad (3)$$

Wh ... ted and ... om figu ... and FR ... 7.6. Thu

Wi ... the bou ...) = 19. ...)m) (t/c ... is cor ... 004 and ... 1 is nea ... ing C_{Do}

Horizontal stabilizer: Reynolds Number = ρVL/μ= 7.593*10^4, so the boundary layer is laminar. Where, Characteristic length (L) =11.09cm. The Re = 7.593*10^4, so laminar boundary layer. For tail C_{Do} expression is of the wing. For New airfoil some values cannot be estimated. So, the value of FFht = 1, CF = 0.005, and horizontal tail CDo = 0.01.

Vertical Tail: Reynolds Number (Re) =ρVL/μ= 4.361*10^4, so the boundary layer is laminar. Where, Characteristic length (L) =6.37cm Thus, CF = 0.006, FFvt = 1. These leads to vertical tail CDo = 0.012.

Landing Gear: It is found that for wheel with one strut, C_{Do} = 1.01 considering front surface area[20]. In a tricycle gear (diameter of 4.8cm, 1.75cm wide main wheels; and diameter of 4.8 cm , 1.75 cm wide nose wheel) total landing gear C_{Do} = (2)(1.01)(8.4)/1843.2 + (1)(1.01)(8.4)/1843.2 = 0.01381 based upon the wing plan form area.

Propulsion unit: It is found that for powerplant, C_{Do} = 0.34 considering front surface area[20]. In 5.515cm^2. frontal area the propulsion C_{Do} = (0.34) (5.515)/1843.2 = 0.00101 considering wing area.

Overall C_{Do}: Overall C_{Do} can be calculated from total of components. So, overall CDo=0.0138+0.0145+0.01+0.012+ 0.01381+0.001= 0.06511 considering wing area of 1843.2cm^2.

Total drag expression: Previously calculated values are
e =0.99, k= 0.0664; C_{LO}= 0.4
The drag polar equation[19]

$$C_D = C_{Do} + \frac{C_L^2}{\pi\, AR\, e} + k(C_L - C_{LO})^2$$

Becomes $C_D = 0.06511 + 0.0643\ C_L^2 + 0.0664\ (C_L - 0.4)^2$, Above polar equation can be used to estimate drag of the aircraft. For level flight at 10m/s at 5 degree AOA, C_L=0.602 from XFLR5 analysis. Thus, using above polar C_D is calculated i.e. 0.09128.

Recommendation: Further enhancement of this research is important to implement it in service of mankind. Thus, modification need to be carried with model to build a stable and easily maneuverable aircraft. Styrofoam locally available isn't appropriate from strength point of view. Thus, balsa wood should be used as replacement of Styrofoam for better strength. Pusher configuration would be better for safety of propulsion unit and flying wing model would be better from stability point of view compared with another model.

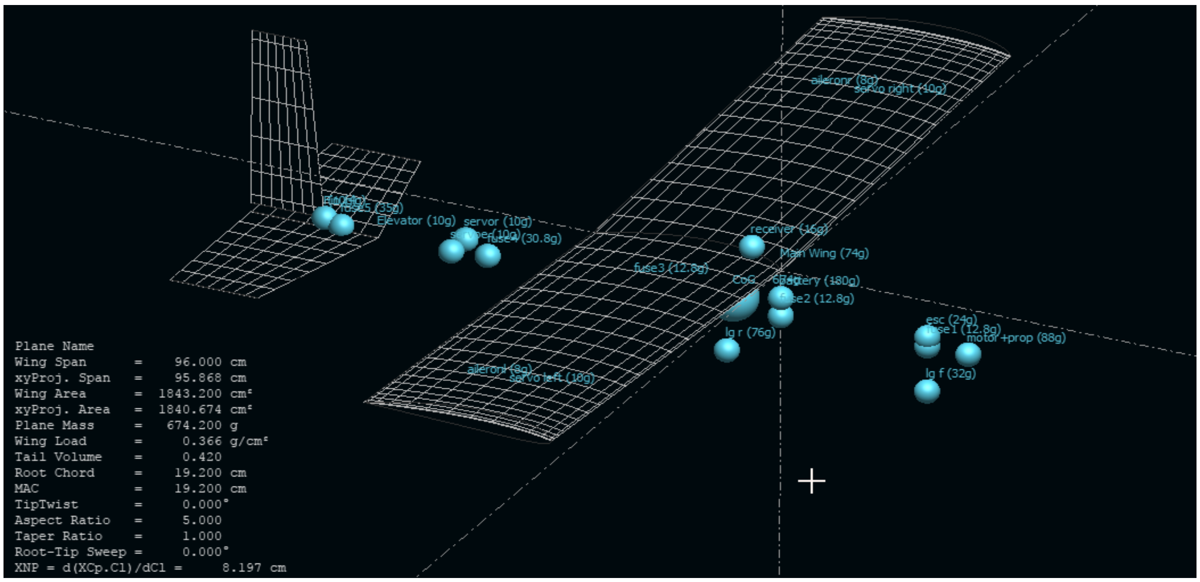

Figure-9: Mass distribution (XFLR5)[6].

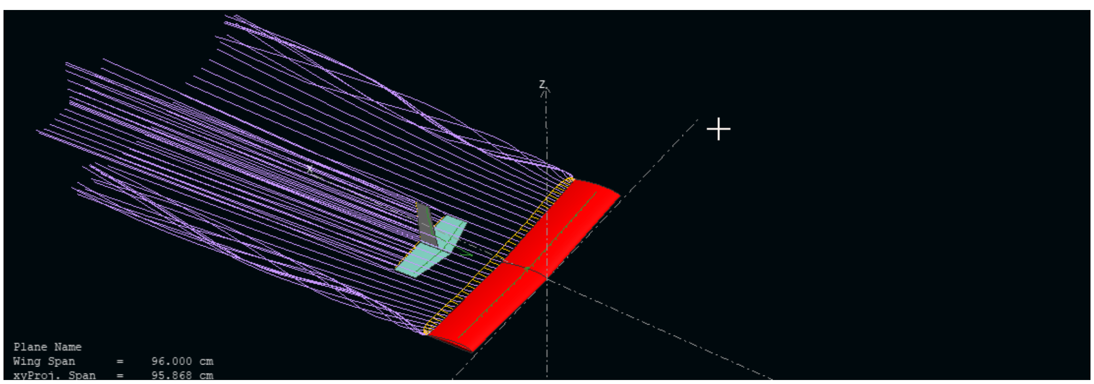

Figure-10: Analysis of wing by VLM1 (XFLR5)[6].

Acknowledgement

I would like to deeply express my gratitude to Er. Bharosh Kumar Yadav, Department of Mechanical Engineering for unyielding support and advice. I would also like to express my gratitude to Er. Anu Shrestha, HOD of Mechanical Engineering and Er. Lal Babu Prasad, Deputy Head of Mechanical Engineering, IOE Purwanchal Campus for their support and guidance. I would also like to thank Apil Acharya, Daya Ram Khadka, Himal Gautam, Santosh Katwal, Saroj Bista, Satish Tiwari and Sohel Alam who were my project team member on project titled "Design, Modelling and Fabrication cum Assembling Powerplant of Fixed wing RC Aircraft."

References

1. Kaisar M.S., Uddin N. and Mashud M. (2015). Design and Construction of a Radio Controlled Airplane by using Delta Wing. *Souvenir from International Conference of Mechanical Engineering and Renewable energy*, 26th-29th Nov.

2. Krishnamurthy S., Jayashankar S., Rao S.V., T.S.R.K. and Nyamannavar S. (2014). CFD Analysis of an RC Aircraft

Wing. *International Journal of Mechanical And Production Engineering*, 2, 63-68.

3. Raj R., Mishra S.K. and Shrivastava S. (2014). Design of a RC Aircraft. JSS Academy of Technical Education, Noida.

4. Bailey J.N. (2010). Design of a low cost powered R/C Combat Airplane and Manufacturing Plan. California Polytechnic State University, San Luis Obispo.

5. Sadraey Mohammad H. (2013). Aircraft Design:A Systems Engineering Approach. A John Wiley & Sons Ltd, USA, 19-47. ISBN 978-1-119-95340-1

6. Andre Deperrois. XFLR5. (Version 6.43) [Software]. Available from:http://www.xflr5.com/xflr5.htm.2003

7. Pamadi Bandu N. (1998). Performance, Stability, Dynamics and Control of Airplanes, American Institute of Aeronautics and Astronautics. United States of America, 67-110. ISBN: 1-56347-222-8

8. Nakayama Y. (1998). Introduction to Fluid Mechanics. Yokendo Co. Ltd, Japan, 175-176.

9. Anderson John D. (2001). Fundamental of Aerodynamics. The McGraw-Hill Company, New York, USA, ISBN: 0-07-237335-0

10. Dutta Soumya (2008). Design/Build/Fly the Evolution of a Model Airplane. University of Tennessee Honors Thesis Projects, Knoxville.

11. Lennon Andy (1996). Basic of R/C Aircraft Design. Air Age Inc., Ridgefield, USA, ISBN: 0-911295-40-2

12. Niu Chung-Yung Michael (1988). Airframe Structural Design. Conmilit Press Limited, California, USA, ISBN: 962-7128-04-X

13. Quora (2018). How come low aspect ratio wings stall at a higher angle of attack?. https://www.quora.com/How-come-low-aspect-ratio-wings-stall-at-a-higher-angle-of-attack. 11/11/2018

14. Pradyumna S.V., Prateek R. and Raju B.S. (2015). Design, Fabrication & Analysis of a High Winger Conventional Tail Radio. *International Journal of Engineering Research and General Science*, 3, 900-921.

15. Rcplane Online (2018). Trainer Design. https://rcplanes.online/ design.htm. 12/11/2018

16. Ansys Inc. ANSYS, (Version 2015) [Software]. Available from:https://www.ansys.com/products/all-products.2015

17. M.H. Sub I. Wattflyer. http://www.wattflyer.com/forums/showthread.php?t=61448. 14/11/2018

18. Singhal A., Khumbhkar T., Sadegaonkar A.S., Krishna A., Sreenath A., Zaare S.A., Gautam R. and Pimpalshende V.R. (2016). Design Report of a Radio Controlled Heavier than air model aircraft. Researchgate. https://www.researchgate.net/publication/295402099.

19. Capable Computing Inc. MotoCALC (Version 8) [Software]. Available from:http://www.motocalc.com/motodown.htm. 2018

20. Anderson J.D. (2010). Aircraft Performance and Design. Tata McGraw-Hill, India, 51-129. ISBN:0-07-00197-1.