TechTatva '19
Embracing Contraries

Sky Rush

MANIPAL INSTITUTE OF TECHNOLOGY
NATIONAL AEROMODELLING COMPETITION

Design Report of an electric-BLDC-powered, propeller driven, radio controlled, heavier-than air aerial vehicle

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1. Project Abstract

The following document is a design report that gives to sufficient detail the necessary rationale behind the decisions taken in relation to design of a micro-class aerial vehicle keeping in mind the constraints imposed by the organizers of the competition. These include constraints on the wingspan, the thrust-to-weight ratio of the aircraft and other restrictions on the use of electronics to monitor the maneuverability of the aircraft. The challenge thus imposed required the design of a plane capable of executing maneuvers while being stable during cruise.

2. Design constraints

The document containing the details of the competition clearly states the maximum thrust-weight ratio of the plane must not exceed 1. The wingspan of the plane is limited to 120cm. Other constraints placed involve the restriction the use of electronics like gyros and IMUs for correcting the motion of the aircraft during flight. Also payload release mechanism must be such that it releases the payload as a single integral unit. No calculations were performed in relation to the take-off of the plane as it is hand-launched.

3. Velocity estimate of the plane

One of the more basic aspects of the design of the plane includes estimating the flow characteristics around the airfoil of the wing. This is one of the first steps we chose to work on because airfoil lift drag characteristics vary with **Reynold's number** of flow around it. The behavior of an airfoil is strictly dictated by the Reynold's number of flow in which it operates. Hence an estimate on the velocity of the plane will give us the type of flow around any airfoil we choose for a given chord. Reynold's number for an airfoil is defined as:

$$R_{e,airfoil} = \frac{\rho v C}{\mu}$$

Where C is the chord of the airfoil; v is the velocity of flow around the foil; μ and ρ are the dynamic viscosity and density of the air at the flight temperature respectively. Assuming flight altitude doesn't vary significantly to cause variations in density and kinematic viscosity (the ratio of density and dynamic viscosity), their values fix at 1.225 kg/m^3 and 1.655 x 10^{-5} Poise. Parameters that are unknown and plane specific are the velocity and the chord. The velocity here is essentially the plane's velocity along with the free-stream flow. The latter is subject to local weather conditions and is not amenable for control under design. Hence a zero wind weather condition was assumed. A reasonable estimate on the velocity of the plane will enable us to tweak the Reynold's number of flow to a desirable range where the airfoil operates in a region to sufficiently overcome the weight of the plane at small angles of attack.

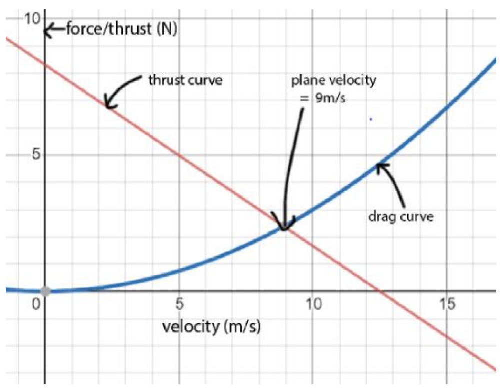
What imparts velocity to the plane is the thrust and thrust is provided by the propulsion system. The propulsion system consists of the motor and the propeller blades. Propeller blades are essentially airfoils stacked adjacent to each other. Hence we approached the problem mathematically by modeling a blade element of the propeller at 70 percent of the radius of each blade as done most often in propeller blade element theories.

Consider the **blade element at 70 percent of the radius of the propeller blade**. For all RC propellers the angle of attack for a given velocity is ensured constant for all airfoil sections by providing a constant pitch angle. Hence corresponding to a given flow velocity there is a definite angle of attack for all sections. This can be found out by employing simple trigonometry.

The net aerodynamic force this section generates is resolved as the thrust force it provides and the force responsible for creating the counter-torque and deciding the RPM of the motor. The equation for the thrust therefore is calculated by applying the basic airfoil lift equation and resolving forces.

$$T(v,\omega) = \frac{1}{2}\rho A(0.7R\omega)(\sqrt{(0.7R\omega)^2 + v^2})\left(C_{L_o} + C_{L_\alpha}\left(\theta_p - \tan^{-1}\left(\frac{v}{0.7R\omega}\right)\right)\right)$$

In simple terms it is the basic lift equation applied to a propeller blade considering appropriate airflow. Here R is radius of the propeller blade and ω is the RPM of the motor. Hence a decision on the type of motor we used was to be taken. **As an estimate we used a BLDC 1200kV motor powered by 11.7 V LiPo batteries.** We measured the RPM of the propeller blades at normalized



The thrust and drag curves plotted in desmos

cruise **throttle of 75%** using a **tachometer** as **5650**. **The pitch angle of most commercial RC propellers is around 7 degrees.** Fixing these data, the thrust as a function of velocity was plotted. A generalized drag function of velocity was generated by taking previous year data of drag encountered for similar planes. The point where both curves intersect is the point where the aircraft should ideally cruise at. This was obtained at **9m/s** for the given specs.

4. Wing design

The wing is the most active lift producing component in the plane and by extension, the most important component that ensures our plane is capable of flight. We wanted a design of the wing that could help us maximize our payload fraction, maximize lift and minimize drag. The two most important features of the wing are its geometry and the airfoil, the parameters required for unambiguously defining a wing and ensure ease of fabrication. While designing the geometry of the wing we took into account the span, aspect ratio and the planform of the wing.

4.1 *Span:*

Going by the constraints imposed by the organizers, we decided on getting our span right to avoid crossing the rules. We set our span at 115 cm to ensure that we got maximum lift while not getting too close to the given span limit of 120 cm. We wanted to maximize the span as the increase in lift provided by a wing with a larger span overshadowed the gain in weight. This allowed us to increase the payload fraction of the aircraft. The large span of the wings also allowed us to reach the required amount of lift at lower velocities and at different angles of attack, hence giving us a reasonable resistance to environmental changes that could cause changes in effective angle of attack or reduce the relative velocity without resulting in a crash. The payload fraction of an aircraft in relation to the span can be obtained by applying the definition of the quantity and simply taking cruise conditions:

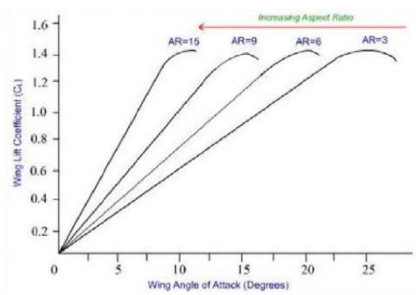
$$P.F = \eta = \frac{\left(\frac{1}{2}\rho v^2 C_L - \frac{W_r}{s^2}AR - \varphi\right)}{\frac{1}{2}\rho v^2 C_L}$$

Where W_r denotes the weight of the plane excluding the wings and φ is the areal density of the wing. The rest of the symbols have their usual meaning.

From the above equation we can conclude that payload fraction increases with increasing span and it's in our favor to maximize the span especially when there is no weight cap to get a better payload fraction.

4.2 *Aspect ratio*

Aspect ratio of the wing is defined as the ratio of the square of its span to its area. It determines several properties of a wing such as the 3-D lift coefficient slope of a wing, induced drag and the



Wing lift vs. Aspect ratio

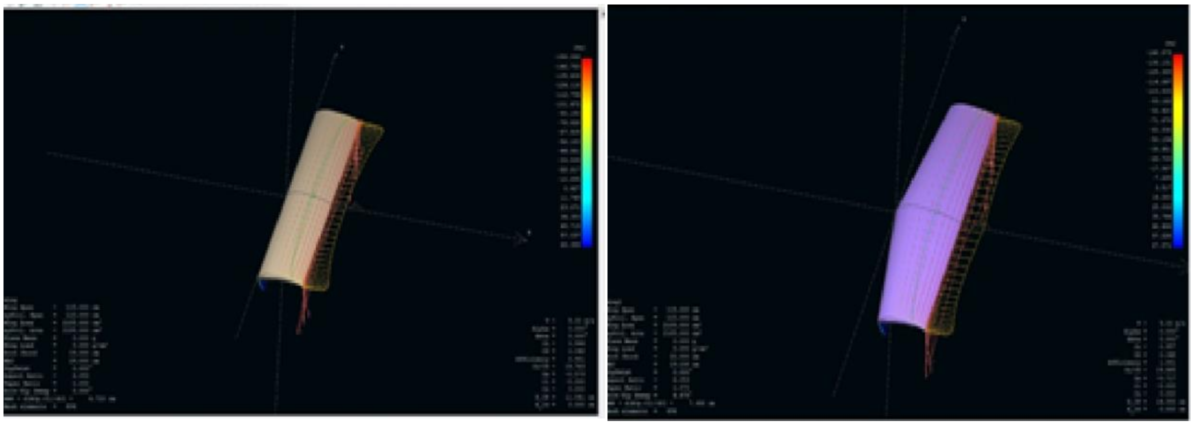
glide ratio of the plane. The aspect ratio also affects the lift generated by the wing for a constant span. We know for a given span, lift will increase with decreasing aspect ratio, so a very high aspect ratio would not produce much lift for a given span. Also if the aspect ratio is very high, the resultant chord will be small, which would reduce the Reynolds number of flow over the airfoil, giving us undesirable flow characteristics. On the other hand a very low aspect ratio will produce lift but will also result in high induced drag which will lower the plane's velocity and will end up decreasing the total lift that the wing generates. Hence an intermediate value of aspect ratio is necessary to ensure adequate lift and ensure proper flow around the airfoil. For our plane we took the aspect ratio at an intermediate value of 6 following the graph above.

4.3 *Planform*

Planform of the wing is the shape taken by the wing when the wing is viewed from above. The planform of the wing determines the wing efficiency and will affect the induced drag produced for a given span and aspect ratio. Mathematically it is expressed in the equation:

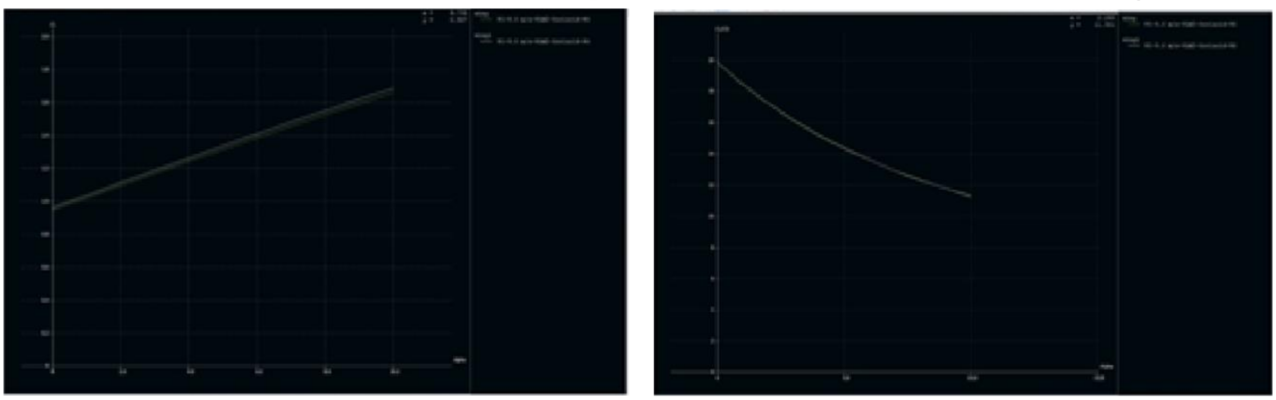
$$C_{D-induced} = \frac{C_L^2}{\pi e AR}$$

Here e is the planform efficiency factor. The wing planform that has the best efficiency is the elliptical wing but it's hard to manufacture. A rectangular wing on the other hand is easy to manufacture but is not very efficient. The middle ground is a tapered wing whose efficiency and



XFLR5 Analysis comparing rectangular and tapered wing planforms

ease of manufacture is between the other 2 planforms. Hence for reduced induced drag while still

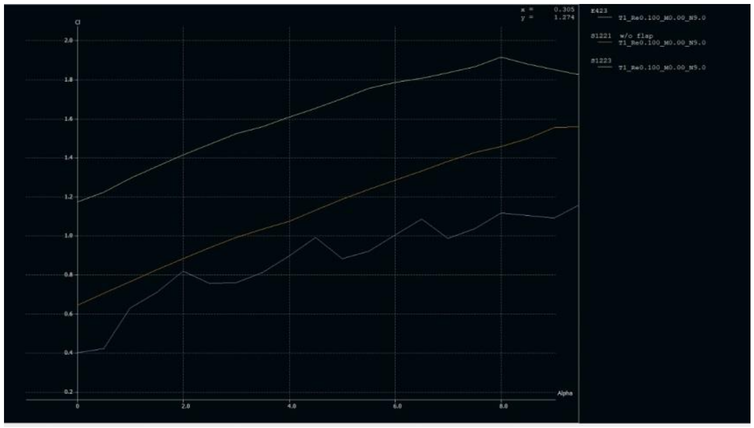


Graphs of lift and wing efficiency vs. angle of attack for the two wing planforms keeping the wing easy to manufacture we have made our wing into a tapered wing. Then a proper taper ratio was defined for the wing, keeping in mind the minimum Reynolds number of flow we wanted while deciding the length for the length of the tip chord and then the root chord was made the appropriate length to maintain the aspect ratio we wanted. The wing was later analyzed in Ansys Fluent for its lift drag etc. The drag curves data was used to calculate the velocity for the second iteration. The result of the second iteration didn't alter much the initial estimate.

4.4 *Airfoil selection*

Airfoil selection is amongst the most important parts of designing the wing. The airfoil shape is what gives lifting characteristics to the surface of the wing. The airfoil determines the lift generated by the airfoil, its inherent pitching moment and the angles of stall for the wing. The airfoil that gives high lift, has a high angle of stall and a roughly constant pitching moment is the airfoil we will choose. Ideally these are the ones with high camber.

We initially consider airfoils that will have favorable characteristics in the conditions at which our plane will operate i.e. low speed, low Reynolds number conditions. Using this, we narrowed



C_L-α curves for the various airfoils we narrowed down to

the number of airfoils down to 3, namely e423, s1221 and s1223. Out of these s1223 had the most favorable lift characteristics, whereas the other factors were similar amongst the 3. Hence s1223 was the airfoil we chose.

5. Empennage design

An aircraft has three axes about which its motion can be analyzed. These are the longitudinal, lateral and directional axes. Two of these namely the horizontal stabilizer and the rudder are responsible for the stability of the plane in the longitudinal and directional axes respectively. Both of these are attached at the empennage for most conventional aircraft. The horizontal tail provides longitudinal stability and controls pitching moment of aircraft. The sensitivity of pitch is determined by horizontal tail volume coefficient given by:

$$V_H = \frac{S_H l_H}{SC}$$

Where, S_H=horizontal tail area, L_H=horizontal tail moment arm, S=reference area (wing area), c=wing chord (mean aerodynamic chord). A well-behaved aircraft has V_H in the range of 0.3 to 0.6, L_H at 55% of fuselage length which gives us 42 cm. Taking V_H as 0.3 we get area of horizontal stabilizer as 300 cm^2. Hence the sum of root and tip chord, C_{root}+$C_{tip,}$ is 20. For easy movement of elevator, we take C_{tip}=6 and hence C_{root}=14. As per a document put up bu IIT Madras as a part of an NPTEL course, the aspect ratio of horizontal tail should be in range of 3 to 5 and taper ratio should be 0.3 to 0.6. Therefore, taking aspect ratio=3 we get span of horizontal stabilizer to be 30 cm. The vertical tail provides yaw damping and hence stabilizing the directional oscillations of the aircraft. Its effectiveness is controlled by vertical tail volume coefficient given by:

$$V_T = \frac{S_T l_T}{SC}$$

Where S_v=vertical tail area, L_v=vertical tail moment arm, S= wing area, b=wing span. Most well behaved aircraft have V_v in range 0.02 to 0.05. A T-Tail specification should have the height of vertical tail somewhere around 50% of horizontal span (as a reasonable approximation) which gives us its height as 15 cm. For a T-Tail the aspect ratio should be 0.7 to 1.2. Taking it as 1.2 we get area of vertical stabilizer as 187.5 cm^2. Taper ratio should be 0.7 to 1.2. Taking it as 0.6 we get C_r=15.6cm and C_t=10cm. L_v= 52.5-9+(16-0.75*13.23) -(22-0.75*19.16) =41.94. By substituting all values in above equation, we get value of V_v as 0.0325 which fall in the range and is suitable for use in our aircraft.

6. Fuselage design

Our fuselage is made up of **Depron** and reinforced by **aeroply and balsa**, as reinforcing of complete fuselage will make it heavy, we cut parts in the middle of the aeroply by doing topology optimization at less stress concentrated zones. The fuselage includes the payload bay within it and the payload is actuated by slider crank mechanism which fits exactly under the wing and the placement of payload is done in **a such a way that the cg change after the fuselage is not substantial.**

7. Longitudinal stability

Albeit we require a maneuverable plane, the stability of the plane can be altered to various degrees by positioning the COG and neutral point of the plane accordingly. To provide an analytical basis to the stability of the plane and look at the effects of various parameters that influence the plane stability along the longitudinal axis, a generalized moment equation about the COG of the plane was used. A trim of 5 degrees was used for the wing to get the desired lift for the aircraft to balance its weight. This was then altered by converting all terms of the equation into coefficients instead of forces. Required data was plugged in from XFLR 5 analysis of the wing. The simplified equation reads:

$$C_m = C_{mac} + (C_L)\frac{l_{cg}}{c} - V_H\left(C_{L_E}\right)$$

Expanding the terms within the stall region:

$$C_m = C_{mac} + \left(C_{L_{op}} + s_p(\alpha+5) + \frac{A_t}{A_w}\left(C_{L_{oe}} + s_e(\alpha + t + \varepsilon_0 cos\beta)\right)\right)\frac{l_{cg}}{c} - V_H\left(C_{L_{oe}} + s_e(\alpha + t + \varepsilon_0 cos\beta)\right)$$

The various symbols mean the following:

C_m : The moment coefficient of the plane.

C_{mac}: The moment of the wing of the plane about the mean aerodynamic chord

C_{Lop}: The lift coefficient of the plane at zero angle of attack

S_p: The slope of the lift coefficient-alpha curve for the plane

α: Angle of attack without trim

5: trim setting for the wing

A_t: Area of the elevator

A_w: Area of the wing

C_{Loe}: Lift coefficient of the elevator at zero angle of attack

S_e: slope of the elevator lift coefficient-angle of attack graph

t: Trim setting of the elevator

ε_0: downwash angle for the elevator placed in line with the wing

β: Elevation angle of the elevator from the MAC of the wing for a T-tail

l_{cg}: Distance of the COG of the plane from the MAC

V_H: Volume coefficient of the horizontal stabilizer = $S_H l_H/SC$; with l_H and S_H denoting the distance of the horizontal stabilizer from the MAC of the wing and the total area of the elevator respectively.

Differentiating the simplified equation above and setting it less than zero for longitudinal stability, we arrive at:

$$l_{cg} \leq c\,\frac{s_t}{s_w}\left(1-\frac{\partial\varepsilon}{\partial\alpha}\right)V_H$$

At neutral point:

$$l_{np} = \ c\,\frac{s_t}{s_w}\left(1-\frac{\partial\varepsilon}{\partial\alpha}\right)V_H$$

According to an NPTEL Document on empennage sizing V_H is fixed at 0.3 for stabile planes. The slopes can be gotten readily from the graphs of XFLR5. The chord is fixed at 19.16 (Mean Aerodynamic Chord). To evaluate the downwash effects, consider the following equation:

$$C_{L_{plane}} = \ C_{L_{wing}} \ + \ C_{L_{elevator}}.\frac{A_w}{A_E}$$

Differentiating:

$$\left(1-\frac{\partial\varepsilon}{\partial\alpha}\right) = \ \left[\frac{C_{L_{\alpha_p}} - C_{L_{\alpha_w}}}{C_{L_{\alpha_E}}}\right]\frac{A_w}{A_E}$$

$C_{L_{\alpha_p}}$denotes the derivative of the lift coefficient for the plane and the same for the other notation used in the above equation. Thus, plugging in data into this from XFLR5 and using the result in the equation for neutral point, we get theoretically the value of the neutral point as measured from the leading edge of the wing (as taken from the MAC) as **9.54 unit**s

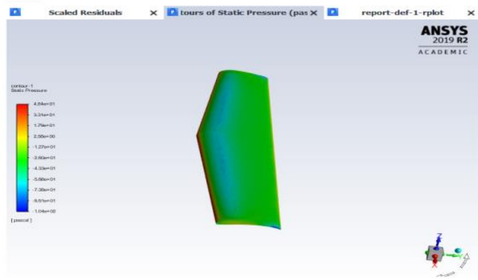
This now allows examining the degree of stability of the plane. Using the reference of a document by IIT Madras, the degree of stability depends on the magnitude of a parameter ‘SM’ (Short for Static Margin) defined as:

$$SM = \frac{h}{c} = \frac{x_{np} - x_{cg}}{c}$$

For stable planes the value of SM should be between 0.05 and 0.25, with the stability getting smaller with smaller values of the static margin. For this competition that requires the plane to execute limbos, a weakly stable plane with quick response to the elevator flap is preferred. For the plane with neutral point at 9.54 units from we have a SM of 0.09. This is quite reasonable as it falls on the mid-range side of the prescribed values-neither too stable to resist pitching when elevators are pulled-up or down by servos or too unstable to get into haphazard phugoids.

The longitudinal stability analysis was performed directly using XFLR5 and the results obtained showed the neutral point location at **9.87 units.** The closeness of these values indicated the accuracy of the theoretical model used for the longitudinal stability of the aircraft. This is also an indication that the software considers the downwash effects for a T-tail configuration as well.

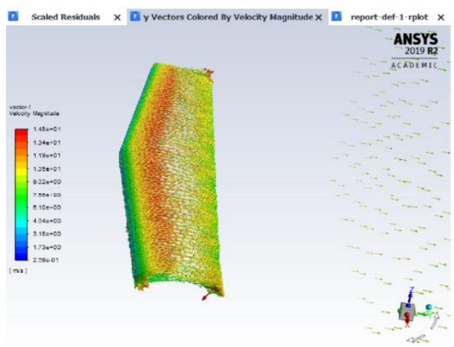
An investigation on the difference in values that XFLR5 throws and what the equation predicts was done. The equation of the neutral point was evaluated in 4 cases: 2 that included downwash



Static Pressure contours for our wing

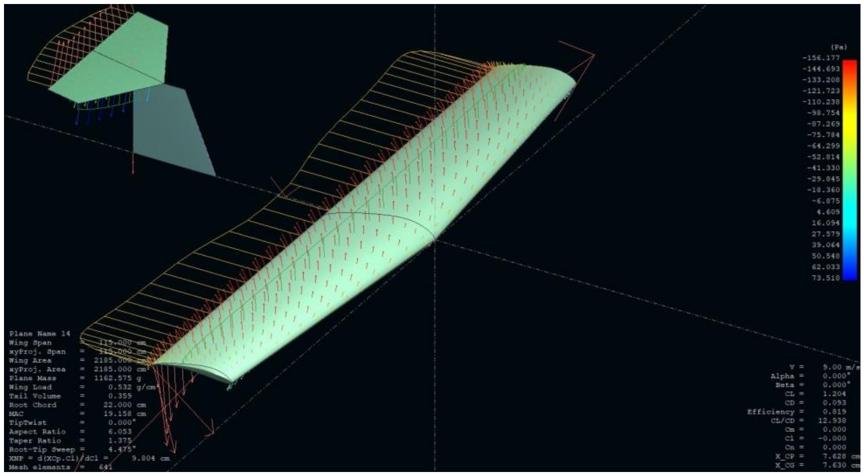
and 2 that didn't, and 2 that included the exact slope ratio and 2 that approximated the slope ratio to 0.5 according to the Piercy approximation (Source: Wikipedia).

In the four cases mentioned above, the ones that didn't include downwash showed significantly



Streak lines of the flow over our wing

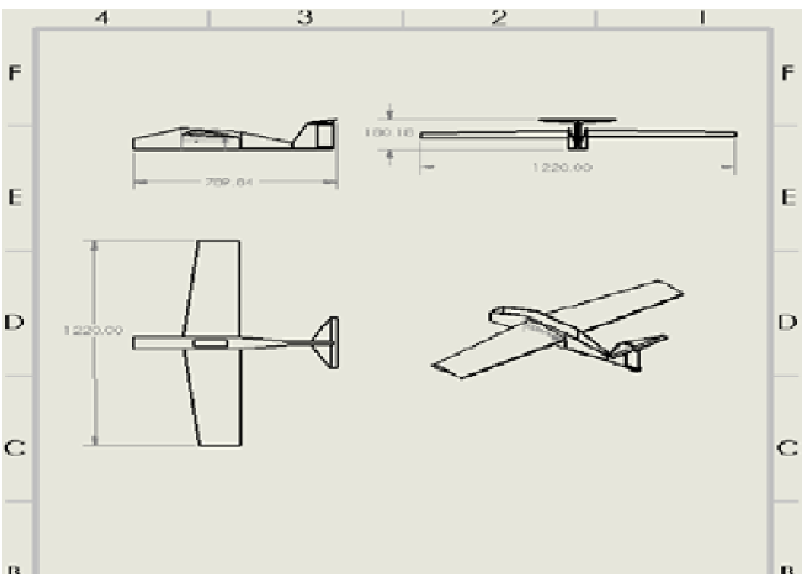
different values. However the ones that had downwash effects showed closeness of values. Nonetheless, the approximation that included the slope ratio as 0.5 showed close resemblance to the value that XFLR5 throws. Hence the corrected static margin for our plane was 0.15-close to the stable region but nonetheless sufficiently unstable. Also one can get C_m-α curve for the aircraft approximately from the equation given above. We want our plane to cruise and stabilize at a pitch angle of zero degrees so that the angle of attack of the wing is equal to the trim setting we put (which is 5 degrees). Therefore the curve must pass through the origin. Setting values of C_m from XFLR at $\alpha=0$ for a zero trim in the expanded equation we get a resultant downwash angle as -8.3 degrees. Note that for all practical purposes we use the value of l_{cg} as zero as efforts are usually made to keep the COG close to the Mean Aerodynamic Chord. Hence following ease of fabrication $l_{cg}=0$. Assuming this doesn't change appreciably at a different angle of attack (i.e. the angle β remains constant with angle of attack variation) or a change in the elevator trim, we introduce a trim of't' for which the value of C_m vanishes. Calculating this trim from the equation, we arrive at a ball park trim of **-10 degrees**. Tweaking the values of trim in



XFLR5 analysis of our T-tail

XFLR5 close to -10 degrees, we arrive at our desired graph at an exact trim of **-9.5degrees.** Nonetheless challenges in fabrication limit our values to -10 or -9 degrees. Fractional values of angles are harder to fabricate and incorporate into designs.

8. Weight build-up



SNo.	Components	Weight (grams)
1	Wing	250
2	Battery	180
3	ESC	30
4	Reciever	10
5	Motor mount	20
6	Motor and props	65
7	Horizontal tail coroplast	11
8	Vertical tail coroplast	15
9	Servos*4	60
10	Payload	120
11	Payload bay	60
12	fuselage	215
	Total	1036

**THE END **