Society of Automotive Engineers

Aero Design Challenge 2020

(SAEISS-ADC 2020)

Design Report of an Electric Motor-Powered Radio Controlled Heavier-than-air Regular class Aircraft.

Submitted by

Team Udyat (ADC2020114)

SASTRA Deemed to be University, Thanjavur

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SAEINDIA
SOUTHERN SECTION

2020 SAEISS AERO DESIGN CHALLENGE

STATEMENT OF COMPLIANCE

CERTIFICATION OF QUALIFICATION

Team Name	TEAM UDYAT
Team ID	ADC2020114
College/University	SASTRA DEEMED TO BE UNIVERSITY
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As Faculty Advisor, I certify that the registered team members are enrolled in collegiate courses. This team has designed, constructed and/or modified the radio-controlled airplane they will use for the SAE Aero Design Challenge 2020 competition, without direct assistance from professional engineers, R/C model experts or pilots, or related professionals.

S.M 18/1/2020

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1. Acknowledgement

‘No man is an island’ said John Donne. Team Udyat is lucky to be connected to many others who keep us from being ‘rifted’. For creativity has no greater sustenance than the love and support of others, we would like to spare this page to acknowledge some of them.

We thank our parents and family members, without whom finishing this report successfully on time would have been strenuous.

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We finally thank almighty for his blessings!

1. Introduction:

Engineering: “The application of scientific principles to practical ends”- *American Heritage Dictionary* This was the motto behind the formation of Team Udyat, a team of seven engineering students from Aerospace and Mechanical Department of the Aeromodelling club at SASTRA(ACS). We decided to utilise the golden opportunity of taking part in the annual Aero Design Challenge 2020 organised by the Society of Automotive Engineers (SAEISS), which we consider as a great platform to practically apply our theoretical knowledge combined with our thought processes.

There are two classes- Regular and Micro. Following an intense discussion and thorough understanding of the rules, Team Udyat decided to take part in the Regular class. As we had experience only in making micro class vehicles, the main objective behind choosing the Regular class was to challenge our fear of making a huge aircraft with good flight performance that adhere to the rule book prescribed by SAEISS. Here, I would like to proudly add that our college seniors, Team BALLISTIC took part in SAEISS ADC 2017 and won overall 2nd. Their guidance and moral support gave us immense confidence to initiate this project.

Project Planning:

“We have the aerodynamic knowledge, the structural materials, the powerplants, and the manufacturing capacities to perform any conceivable miracle in aviation. But miracles must be planned, nurtured, and executed with intelligence and hard work”.

- Glenn L. Martin, aviation pioneer and manufacturer, 1954.

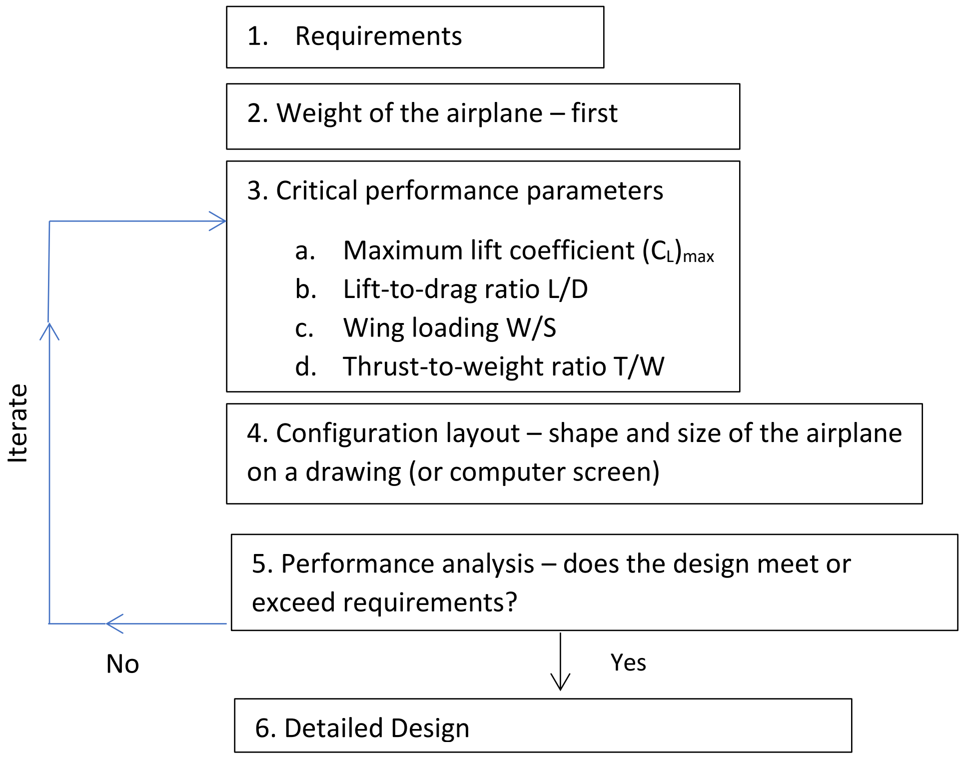
Our initial team meetings mainly consisted of discussions on how to approach this project, as SAEISS not only looks for an airplane design that is certified best for flying, but also assesses the student’s team dynamics, discipline in approaching a problem, and various other skill sets.

2. Design approach:

We decided to split our process of fabricating this aircraft into two phases, namely, Conceptual Design, and Detailed Design. These two covers the web of details that surrounds and interconnects all other parameters that is required for designing an efficient airplane.

2.1 Conceptual Design

We covered 5 pivot parameters under this heading, which is given below as a flow chart.



3.1.1. Requirements and constraints:

- To design an aircraft that can lift as much as weight as possible while observing the available power and aircraft dimensional requirements. Accurately predicting the lifting capacity of the aircraft is an important part of the exercise as prediction bonus points often determine the difference in placement between competing teams.

- Total dimensions of airplane not more than 170 inches.
- Total weight of empty aircraft not less than 2 Kgf and not more than 5 Kgf.
- Use of FRP is prohibited.
- LiPo battery 4 cell – 6 cells.
- Total Take-off distance- 200ft
- Total Landing distance- 400ft.

3.1.2. Weight of the Airplane - First Estimate & Material Selection:

No airplane can get off the ground unless it can produce a lift greater than its weight. Hence, it was important to consider and approximately calculate the gross weight of the aircraft as all the other aerodynamic parameters such as lift, thrust required, span length of the wing, etc. depended on it.

We classified the gross weight in to two - Weight of the payload (W_p) and empty weight of the aircraft (including electric components, W_e).

The primary purpose of this aircraft is to have a stable flight and lift as much payload as possible, for which having a higher payload fraction is essential. Payload fraction is defined as the 'ratio of weight of the payload to the take-off weight of aircraft'. This can be achieved by decreasing the empty weight of the aircraft. Since the minimum weight of the aircraft should be 2Kgf, we considered building an aircraft weighing 2.3Kgf, leaving a clearance of 300g. We were very particular with this consideration as there were high chances of increase in weight upon building. We made sure that the weight didn't exceed 5Kgf. This was also a deciding factor in the selection of material (density). The empty weight of the aircraft depends on the materials used and the electrical components. Intense study and research were done for the selection of material. This was done to quench our thirst in discovering several non-conventional, sustainable materials that can be used to build RC planes. After analysing the pros and cons, cost, availability, ease of manufacturing and the other material parameters mentioned, we decided to go with Aluminium and various grades of Balsa wood. Although these are conventional materials, this activity helped us explore a variety of materials, their manufacturing processes, cost and availability. After going through literatures, we contemplated our payload fraction to be 0.6, which yielded a result that our aircraft can lift a payload of 3.45Kgf.

Conclusion: The gross weight of the aircraft ($W_e + W_p$) = 2.3Kgf + 3.45Kgf = 5.75Kgf = 12.676lb.

3.1.3. Performance parameters:

An estimation of critical performance parameters like C_L max, L/D, T/W, W/S was made as they form the ultimate deciding criteria for calculation of span length, chord, aspect ratio, thrust required, etc.

Maximum Lift Coefficient: This is the stage where we made an initial choice of the airfoil. The aircraft should be able to perform a stable 360º turn. Hence, high speed and manoeuvrability can be neglected. From historical data, Reynolds number for an RC aircraft ranges from 1,50,000 to 2,50,000, and upon choosing an average value of 2,00,000, the average Cruise velocity obtained (V_c) is 9-10m/s (Ref. airfoiltools.com).

Take off speed (V_{TO})= Cruise speed/1.2 = 7.8m/s. Stall Velocity = V_{TO} /1.2 = 5-6m/s.

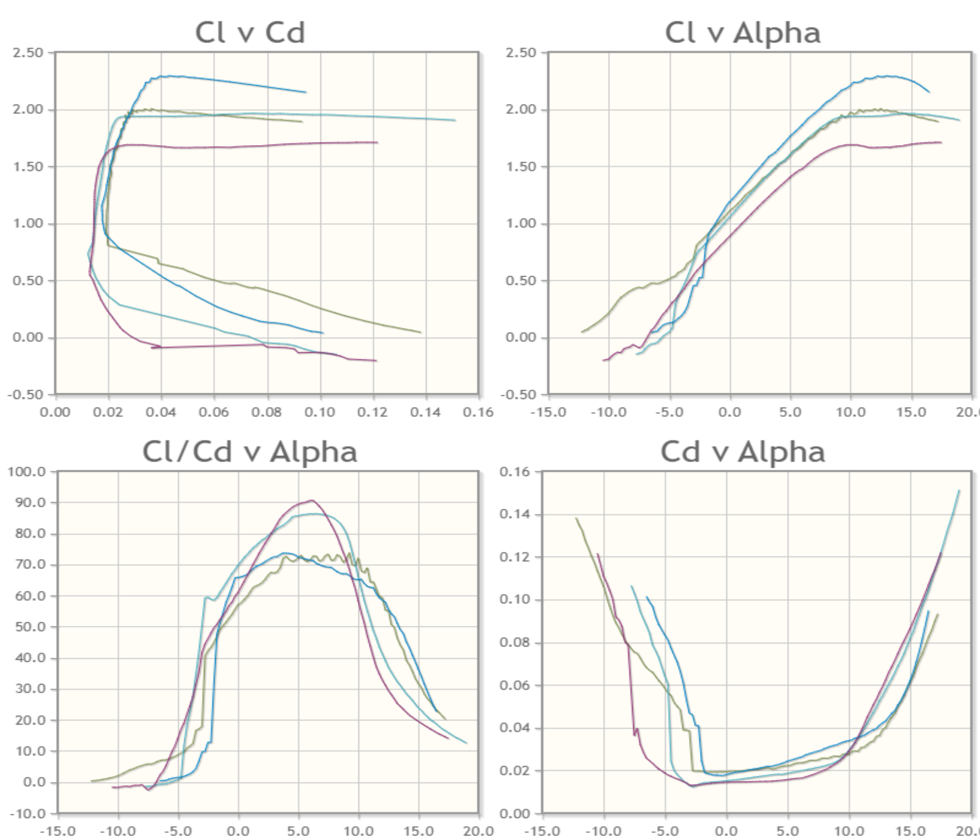
Now, the problem statement is to produce high lift at a relatively low Reynolds number, i.e. 2,00,000. This is when wind turbines drew our attention, whose blade profile is capable enough to produce high-lift in a velocity range of 9-10m/s. Research work made us shortlist the below mentioned airfoils:

- Wortman fx-63-137
- E-423
- S1210
- S1223

Design Objective	Weightage	Wortman	E-423	S-1210	S-1223
C_d min	25%	5	3	5	4
C_{mo}	15%	2	3	3	4
C_{lmax}	15%	3	4	4	5
α_o	10%	3	3	4	4
(C_l/C_d) max	10%	5	2	4	3
$C_{l\alpha}$	5%	2	3	3	4
Stall Quality	20%	2	2	3	5
Summation	100%	3.30	2.85	3.85	4.25

Table.1

Comparatively, the overall performance of S-1223 proved to be better. This was a validation to our gut-feeling that S-1223 would inherently perform well since it is widely used for RC plane (Historical data).



Colour code:

Pale blue: S1210

Blue: S1223

Green: E423

Violet: fx-63-137

Specifications of S-1223:

C_{lmax} @ A.O.A=16^0	2.2919
C_l @ A.O.A= -7.5^0	0
(C_l/C_d) max @ A.O.A =3.75^0	73.64
C_l ideal (C_l ideal is when the corresponding Cd is minimum)	1.1554
C_d min	0.01758
C_l @ A.O.A=0^0	0.01758
(Thickness/chord) max = 12.1/19.8	0.611
Parasite Drag Co-efficient (C_{d0})	0.01758
$dC_l/d\alpha$	5.698

Table.2 source:www.airfoiltools.com

For Wing: C_{Lmaxw}= C_{lmax} * 0.95= 2.177

C_L ideal = C_l ideal *0.95= 1.0973

For Aircraft: C_{LAmax} = 2.177* 0.9=1.95

C_{LA} ideal= 1.0973*0.9= 0.98

Note: The aircraft has to fly at ideal lift co-efficient of 0.98 for it to experience minimum drag.

The gross weight of the aircraft is 5.75 Kgf. At cruise condition, Lift = Weight. Hence, the aircraft should produce a Lift of 5.75*9.81= 56.4075N.

Lift = ½* $\rho * Vc^2 * S * C_{Lmax}$.

56.4075= .5*1.225* 100* S* 1.95

Span area (S)= .4722 m^{2}. = 732 in^{2}.

Wing Loading (W/S): Wing loading is defined as the ratio of gross weight of the aircraft to the span area. The faster an aircraft flies, more is the lift produced at each unit area of wing, since free stream velocity is directly proportional to lift. Hence, even with a small span area, more weight can be lifted just by increasing thrust and thus velocity.

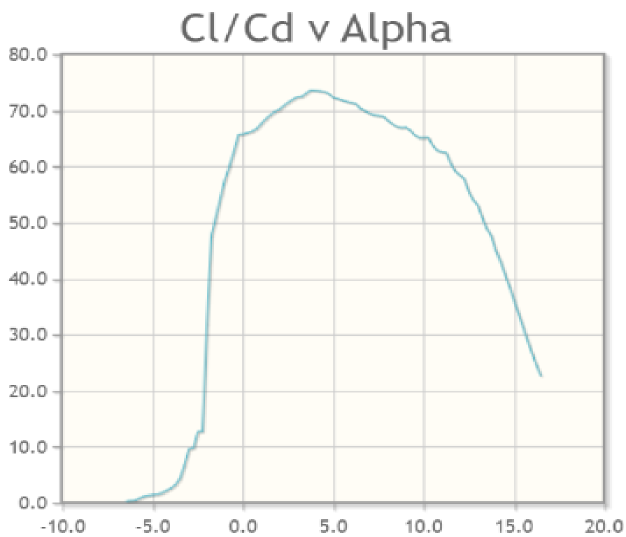
TABLE 1 Model type	Power loading oz./cid 2-stroke	Wing loading oz./sq. ft.	Aspect ratio
High-speed, highly maneuverable	200-250	22 to 26	4 to 6
Moderate-speed sport	250-300	16 to 22	6 to 8
Low-speed trainer	300 and up	12 to 16	8 to 10
Slope gliders	—	12 to 14	8 to 10
Soaring gliders	—	8 to 12	10 to 15

From the table, wing loading range of 16-22 oz/ft^2 is chosen. We arrived at this conclusion because the requirements of a regular class aircraft given by SAE restricts battery usage from 4-cell to 6-cell, which when powers a BLDC motor, generates speed that falls under moderate speed sport category. The corresponding Aspect ratio is 6-8, an average value of 7 is chosen. Aspect ratio = Span length (b) / Span area (S).

Thus, **Span Length (b)= sqrt (732*7) = 71.58 in.**

Mean Aerodynamic Chord (MAC)= Span (b) /Aspect ratio= 10.22 in.

Lift to Drag Ratio (L/D):



From Fig, the $C_l/C_{d\ max}$ ratio for S-1223 is found to be 73.64 at an angle of attack of 3.75. The corresponding C_l and C_d is found to be 1.6172 and 0.02196 respectively. This is only for the airfoil, the L/D_{max} for the entire aircraft is calculated in 3-D drag analysis section using drag polar equation.

Thrust to Weight Ratio (T/W):

The value of T/W determines in part the take-off distance, rate of climb, and maximum velocity. Take-off distance constraint is examined in here.

$$Sg = 1.21 * (\frac{W}{S}) / \left(g * \rho * CLmax * \frac{T}{w}\right)$$

Where, Sg is ground roll distance = 200ft.

W/S is wing- loading, taken as 20oz/ft^2 from wing-loading table given above.

g= acceleration due to gravity= 32.2 ft/s$^{2.}$

C_{Lmax} = 1.95

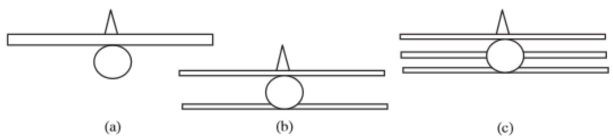
ρ= Density at sea-level= 0.002377 lb/ft^3.

T/W= (1.21*20)/ (32.2*0.002377*1.95*200) = 0.81.

3.1.4. Configuration Layout:

Based on the information that has been calculated so far, the picture of aircraft with dimensions is drawn. This section of intellectual process of airplane design is where intuition, experience and the *art* of airplane design most strongly come in to play. We began with wing design (primary lift producer).

Number of wings-



(a) Monoplane (b)Biplane

(c) Tri-Wing

Monoplane was chosen due to the below-mentioned advantages over any other number of wings.

1) Less weight, Ease of manufacturing and assembly operation.

2) Highest efficiency and lowest drag when compared to other configurations.

Wing vertical location- Feasibility analysis was performed to decide upon the vertical location of the wing. Design objectives were narrowed down and each of it was given a specific weightage and then rated accordingly for every vertical location. High wing has got the highest rating out of all due to many reasons such as-

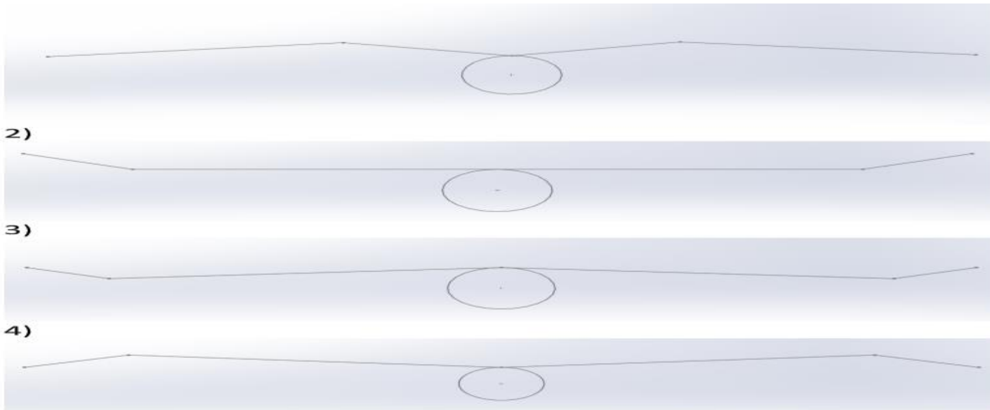
1) Facilitates the loading and unloading of payload into and out of the aircraft.

2) During landing, load on the landing gear will be transferred to the fuselage and very less to the wing thus protecting the wing from harsh landings.

3) It increases the dihedral effect and makes the aircraft laterally more stable.

4) Aerodynamic shape of the fuselage lower section can me made smoother.

3.1.4.1 Wing design- From the primary dimensions of the wing obtained in the previous section such as the chord length, span length and span area, the team had intense brainstorming sessions to decide on the wing



configuration. We came up with 4 different wing designs drawing inspiration from nature which are shown above.

During the decision-making process 8 important parameters had to be considered before arriving at a particular wing configuration. They are:

1) Aspect ratio-

It's the ratio of a wing's length to its chord. Both low aspect ratio and high aspect ratio wing have both advantages and disadvantages so a middle value was taken. **The aspect ratio was chosen as 7.**

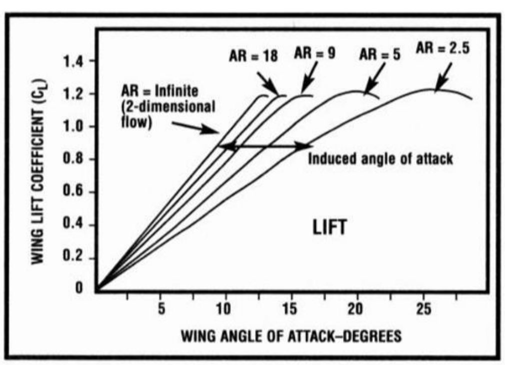


Figure 3.
How aspect ratio affects the stall angle of attack.

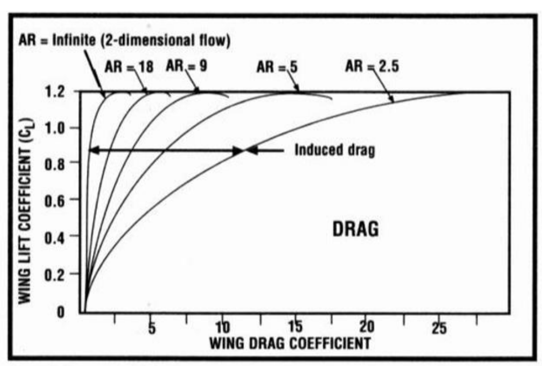


Figure 6.
How aspect ratio affects drag at a given lift.

2) Taper ratio-

The prime purpose of tapering our airplane wing is to attain elliptical lift distribution which has the highest efficiency when compared to any other distribution. One way to attain this is by making the wing planform elliptical which is done by varying the chord length gradually across the span length.

This type of wing though aerodynamically has the best characteristics, is rated the least when the manufacturing aspect is considered.

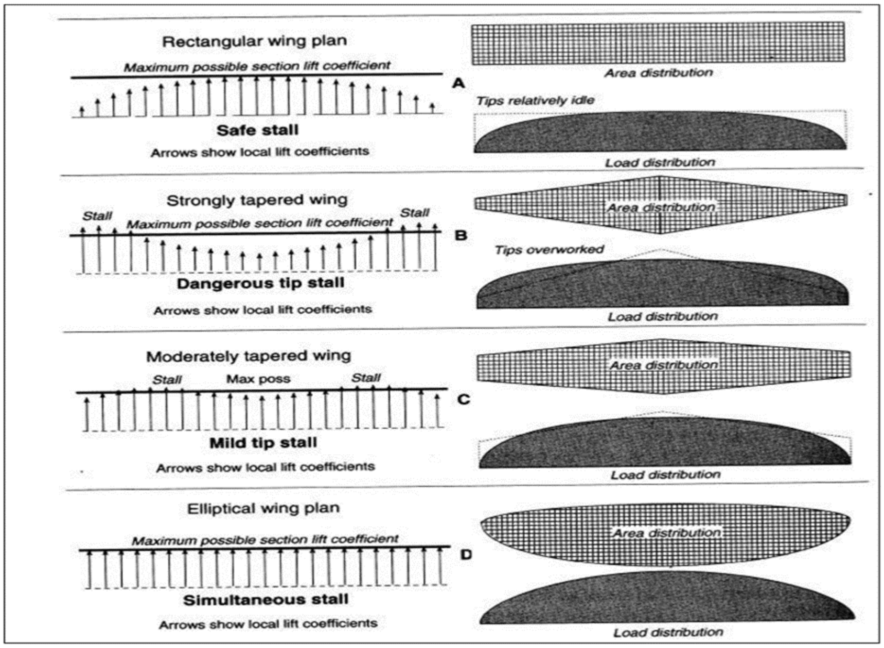


Figure 18 - Stall Onset for Taper Design

When the team was doing literature study to find other ways of attaining elliptical lift distribution, we came across the concept of taper. The lift distribution of a tapered wing is very close to elliptical and the factor that tells us about this closeness is the Oswald span efficiency factor (e). For a tapered wing, this factor lies in the range of 0.90 to 0.98. Also, another factor to consider after deciding to taper the wing is the taper ratio which is defined as the ratio of tip chord to root chord. Taper ratio= C_t / C_r

Optimum range of taper ratio for an RC plane is 0.2 to 0.6. Beyond 0.6 or below 0.2 the wing experiences a phenomenon known as tip stall which is considered extremely dangerous for a plane during flight. We decided the **Taper ratio of our plane to be 0.58.**

3) Sweep angle- Since Sweep angle is used in aircraft which fly in supersonic and high-speed subsonic regions and our plane is designed to fly in low subsonic region, the sweep angle is neglected.

4) Tip chord and Root chord- After deciding the span area and the aspect ratio for a moderate category aircraft, wing's mean aerodynamic chord can be decided using the below mentioned formulas.

$b^2 / S = AR$ $\qquad$ $b / c = AR$

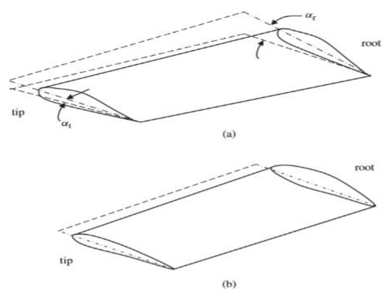
The MAC obtained by the team was 10.1 inches. Root chord and tip chord can be calculated using the below mentioned equation

$$\overline{C} = \frac{2}{3} C_r \left(\frac{1+\lambda+\lambda^2}{1+\lambda} \right)$$

Root chord- 12 inches, Tip chord- 7 inches, MAC- 10.1 inches

5) Dihedral angle-

It is the upward angle from the horizontal of the wings. The purpose of dihedral angle is to provide lateral stability. Dihedral angle 5 degrees was chosen (historical data).



6) Aerodynamic and Mechanical twist- Aerodynamic twist: If the tip airfoil and the root airfoil are not the same, the twist is referred to as aerodynamic twist.

Geometric twist: When the tip incidence angle and the root incidence angle are not same, the twist is referred to as geometric twist.

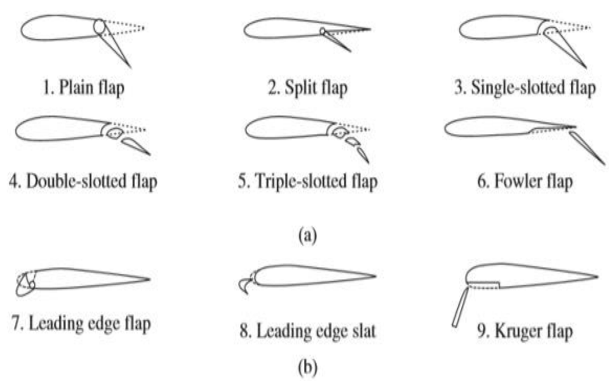
Two major goals for employing a twist (aerodynamic or mechanical) in a wing are-

1) To avoid tip stall

2) To modify the lift distribution into an elliptical one

Even though providing twist can drastically vary the properties of a wing favourable to us, it brings immense difficulties from a manufacturing point of view. Since ease of manufacturing was top priority among all other aspects, we decided not to include a twist in our wing.

7) High lift devices – High Lift Device is a component that increases the amount of lift produced by the wing. The device may be a fixed component, or a movable mechanism which is deployed when required. We have gone with flaps which is the most commonly used high lift device.



Types of Flaps- From the list of flaps mentioned, the team decided to go with **Plain Flap** due to the following reasons- 1) Simplest and earliest type of High lift device. 2) Camber can be increased mechanically which results in an increase in lift. 3) Easiest to build.

8) Wing setting angle –

α= α0 + ((18.24*CL*(1+T))/AR.

α0 is the range of 6-$7^{0.}$ α = 10-11^{0} but α(L=0) = -$7.5^{0.}$

therefore, αsett = 2.5 – 3.5 {No problem of stall}. The αsett = 2.5 to 3.5.

3.1.4.2. Horizontal Stabilizer: The main purpose of horizontal stabilizer is pitch stability.

The airfoil selected is NACA 0009 (Literature study). The tail plane needs to produce both positive lift and negative lift. For this reason, a symmetric airfoil section is a suitable candidate for a horizontal tail. The horizontal tail volume coefficient usually lies in the range of 0.3 to 0.6 for a well-behaved aircraft. If V_h is too small then the aircraft's pitch behaviour becomes sensitive to the CG location and will show poor tendency to gusts. So, it was taken as 0.6.

$$\mathbf{V_h} \equiv \frac{S_h \ell_h}{S c}$$

Where,

S_h = span area of horizontal stabilizer S= Span area of the wing

L_h= horizontal tail moment arm C= Wing mean chord

The L_h is found by using the below formula L_h= Kc*(4*c*S*Vh/π*Df) ^0.5

Kc= correction factor varies between 1 and 1.4

D_f= Effective diameter of the fuselage as L_h=34inches.

Substituting the values for the above parameters, we obtained S_h as 130 $inch^{2.}$

$$AR_h = \frac{2}{3} AR_w$$

Aspect Ratio of horizontal stabilizer was found to be 4.66. We have considered a rectangular horizontal stabilizer. The span length is 24.61 inch the chord is 5.27 inch. Around 40% of the horizontal stabilizer area was allotted for the elevator, resulting chord of the elevator to be 2 inches and the length to be 24.61 inches.

The C_l of the horizontal stabilizer was found to be -0.368. Which is sufficient to counter the moment generated by the wing.

3.1.4.3. Vertical Stabilizer: The main functions of Vertical stabilizer – (1) Directional Stability (2) Directional Trim. Low Aspect ratio to attain high stalling angles. Thus, it was chosen as 2.5.

Vertical tail volume coefficient, Vv =. $\frac{(Sv*Lv)}{Sw*b}$

Where,

Sv: Vertical tail area Lv: Vertical tail moment arm

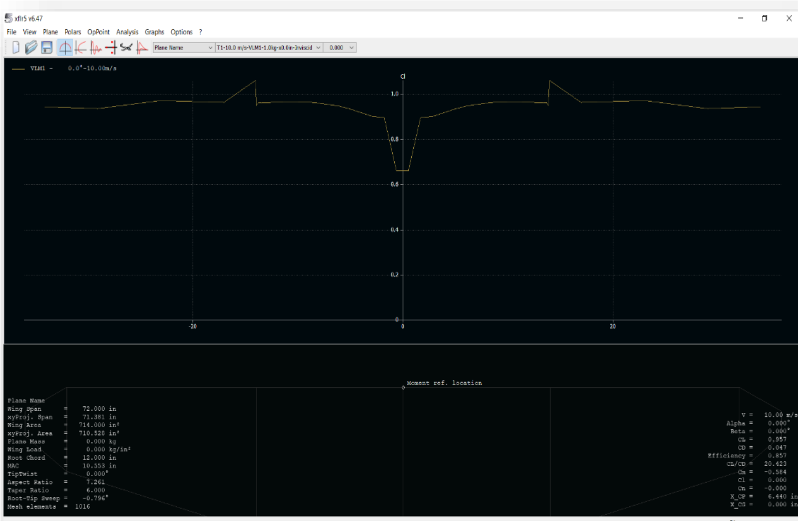
Sw: Wing area b: Wing span

The vertical tail volume coefficient lies in the range of 0.02 - 0.07 (Historical data). Higher the value more the stability. Hence, 0.05 is considered. For an aircraft with a tractor- propeller, the tail arm is about 60% of the fuselage length and was found to be 31.2 inches. The rudder area is generally 40% of vertical stabilizer area. The vertical tail area is found to be 83 $in^{2.}$ Aspect Ratio = 1.55 * h^2 /Vv

Height of the vertical stabilizer is calculated to be 12 inches (formula above). Twin tail configuration is chosen (reduces cantilever stress at the root and less parasite drag); each vertical stabilizer's height is 6 inches with an area of 41.5 in^{2}.

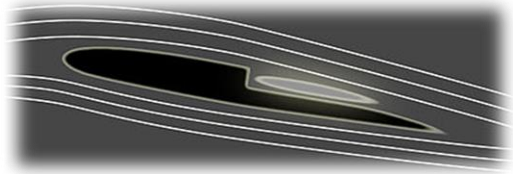
3.1.4.4. Fuselage Construction:

The fuselage of the aircraft is the component to which essentially everything is attached and it must be large enough to accommodate the motor casing, battery and other electrical components in the nose, payloads, ballast compartment (if necessary), provisions for control rods and electrical wires to run through wings and fuselage undisturbed. The overall length of the fuselage is 50 inches, approximately 65% of Span (Historical data) with tapering width and varying height. The rear, empennage section of the fuselage is placed at an inclination of 22.4^{0} with the longitudinal axis to prevent the horizontal stabilizer and elevator being influenced by the wake produced by the main wing. The payload bay's location is designed to coincide with that of Centre of Gravity (C.G)- position of C.G remains the same before and after loading the payload. Material used for the construction is Balsa wood and Aluminium rods (stringers and reinforcements).



Our drag analysis for the 'attached wing and fuselage' design yielded results where there was increase in drag and steep decrease in lift, as expected (picture). A brain-storming session gave us an idea to reduce the slope of reduction in lift. Drag can be reduced if the flow separation is delayed. We tried to draw the principle used in Kline-Fogleman airfoils and applied it to our fuselage.

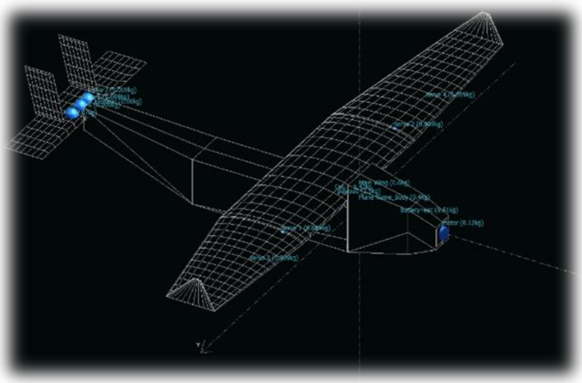
3.1.4.5. Innovation: The Kline–Fogleman airfoil or KF airfoil is a simple airfoil design with single or multiple steps along the length of the wing, designed by Richard Kline and Floyd Fogleman. The



purpose of the step, it is claimed, is to allow some of the displaced air to fall into a pocket behind the step and become part of the airfoil shape as a trapped vortex. This purportedly prevents separation and maintains airflow over the surface of the airfoil. A similar step is made after the diverging part of fuselage to attach the flow and reduce the slope of decrease in lift.

3.1.5. Centre of Gravity and Stability Analysis:

One of the most important characteristics to consider while designing an aircraft is its inherent stability and controllability. Even though evaluating the stability of an aircraft is a complicated process when evaluated computationally, we have alternative analytical methods which give reasonable estimates and



are vastly simpler to apply.

COG of the aircraft was estimated analytically as well as using XFLR5 by placing all the electrical components and payload as point masses.

COG location using XFLR5: ***1.997(from MAC of 10.1 in)***

Analytical method: Effects of moving COG:

Move COG forward: Increased stability => The aircraft is more resistant to AOA and Velocity changes

Move COG backward: Decreased stability => The aircraft is less resistant to AOA and velocity changes.

The COG position that gives neutral stability is defined as the neutral point. (NP)

The degree of pitch stability or instability is traditionally specified by the Stability Margin.

$$SM = (x_{np} - x_{cg})/c$$

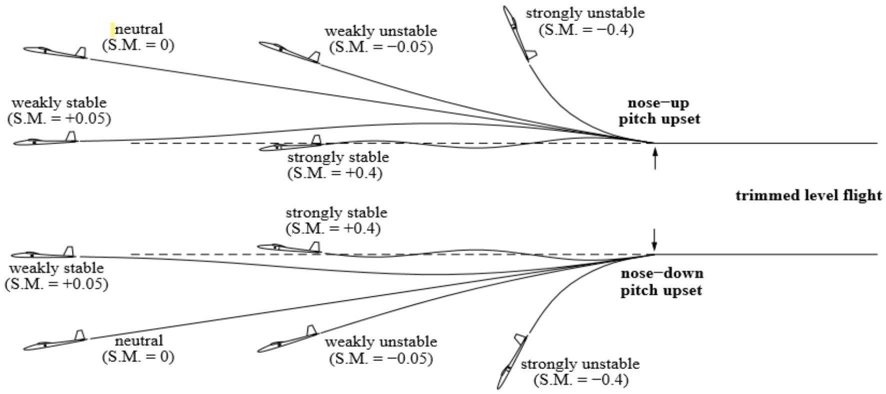


Figure: Natural responses to a pitch disturbance for different amounts of pitch stability.

From Fig, we can determine the optimum stability margin for a pitch disturbance. Stability margin cannot be negative at the same time the value cannot be largely positive due to undesirable effects. One large drawback of a large S.M. is that it causes large (and annoying) pitch trim changes with changing airspeed.

Ideal range of SM obtained from experimental results is

$$\boxed{S.M. \equiv \frac{x_{\mathrm{np}} - x_{\mathrm{cg}}}{c} = +0.05 \ldots + 0.15}$$

Analytical equation to calculate the neutral point of the plane is mentioned below:

$$\frac{x_{\mathrm{np}}}{c} \simeq \frac{1}{4} + \frac{1 + 2/AR}{1 + 2/AR_{\mathrm{h}}} \left(1 - \frac{4}{AR + 2}\right) \mathrm{V_h}$$

X_{np} = 3.027 inches (for 10.1 inches MAC).

We chose the mid value of Stability margin which is 0.10.

X_{cg} = 2.017 inches (from MAC of 10.1 inches).

__3.1.6. Landing gear:__ The main function of landing gear is to absorb and dissipate the kinetic energy of landing impact, thereby reducing the impact loads transmitted to the airframe. Landing gear can be fixed either in fuselage or in main wing. It would be better if we attach the landing gear to the fuselage as it would carry all the load ensuring a safe wing. Though there are roughly 10 configurations of landing gear, analysing the pros and cons narrowed us down to two configuration, tricycle and inverted tricycle. While landing, rear wheel would have initial contact with ground and hence it shall take maximum load. Thus, 2 tyres in the rear to ensure safe landing led us to Tricycle configuration.

3.2. Detailed Design: It is in the detailed design phase that serious structural and control system analysis and design take place. We expected the end of detailed design phase to bring a major change - to commit to the fabrication of plane or not.

3.2.1. 3-D Drag analysis (Analytical) Airfoil Drag polar equation:

$$C_D = C_{d,o} + k\left(C_l - C_{l,0}\right)^2$$

From Table.2, value of k for S-1223 airfoil was found out to be 0.0668 and C_{do} = 0.01758

Wing drag polar equation

$$C_D = C_{D,o} + \frac{C_L^2}{\pi\ AR\ e} + k\left(C_L - C_{l,0}\right)^2$$

$C_{D,o}$ = Parasite drag of wing mainly due to skin-friction and pressure distribution around the airfoil.

2nd term = inviscid drag-due-to-lift 3rd term = viscous drag-due-to-lift

C_D = Drag polar $C_{l,o} = C_l$ for minimum wing drag

Item	Planform area	Wetted area	Reference length
Fuselage	300 sq inch	1200 sq inch(approx)	50 inches
Wing	720 sq inch	1440 sq inch	10.1 inches
Horizontal stabilizer	133.94 sq inch	267 sq inch	5.3 inches
Vertical stabilizer	83 sq inch	160 sq inch	6

$C_{D,O}$ value for wing is practically equal to $C_{d,o}$ (parasite drag fir airfoil) = 0.01758

Aircraft Drag polar equation $$C_D = C_{Do} + \frac{C_L^2}{\pi\, AR\, e} + k\left(C_L - C_{l,0}\right)^2$$

Value of $C_{D,O}$ in the aircraft drag polar equation is different from the $C_{D,O}$ present in wing drag polar equation . C_{Do} results from pressure and skin friction drag due to fuselage, wing, tails, landing gear, engine, etc. Major contributors in the case of an RC plane being the first four, drag due to other components are neglected.

$C_{D,O}$ for Fuselage

Fuselage CDo = FF*Cf*SWet/SRef

FF = form factor representing a pressure drag contribution

Cf = skin friction coefficient of the component (fuselage)

S_{Wet} = wetted area of the component

FF_f = 1 + 60/(FR)^3 + 0.0025 FR

For our model the FR = fuselage fineness ratio = fuselage length/diameter [50/7 = 7.14]

FF= 1.182

$$C_F = \frac{0.455}{\left[\log_{10} Re_L\right]^{2.58}}$$

Cf = 0.0041

CDo = 0.0193

C_{Do} for Wing

Cf= 0.00609

FFw = [1 + (0.6/(x/c)m)*(t/c) + 100(t/c)4]

(x/c)m = maximum t/c location of the airfoil

FF_w = 1.388

CDo = 0.0169.

C_{Do} for Horizontal stabilizer

Horizontal stabilizer- airfoil NACA 0009

Cf = 0.0032 , FF_{hs} = 1.1213, *CDo = 0.007*

CDo for Vertical stabilizer

Cf = 0.0029, FFvt = 1.1213, CDo = 0.00062.

Total $C_{D,O}$ = 0.0193+0.0169+0.007+0.00062 = 0.043

C_D = 0.043 + 0.08599 + 2.27*10^-3 = 0.13127

Therefore, from the analytical 3D drag analysis performed, total drag coefficient for the aircraft is 0.13127.

L/D ratio

$C_{L\ ideal}$ for the aircraft is 1.1554*0.9*0.9 = 0.93

Total C_D = 0.17 (0.4 has been added for the drag contribution of landing gear and other components extruding out of the surface.

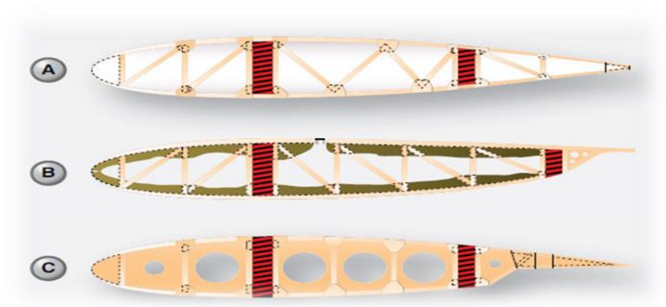
$(L / D)_{max} = (C_L / C_D)_{max}$ = 0.93 / 0.17 = 5.47.

3.2.2. Structural Analysis:

We performed Static Structural Analysis in Solid Works and ANSYS Workbench. This is done to identify the location of spars, internal design of ribs, number of ribs required and distance between ribs.

1) *Location of Spars and Internal Design of ribs:* Spars resist bending and forms wing box for stable torsional resistance. Ribs helps in maintaining the aerodynamic profile of the wing and also ensures the structural integration.

From Literature work, we found 3 widely used internal design whose pictures are given below:



We used the basic bending equation, which is

$$\frac{M}{I} = \frac{\sigma}{Y} = \frac{E}{R}.$$

Where, M= Bending moment, I= Moment of Inertia (obtained from solid works), σ= Yield stress, Y= Distance from the neutral axis, E= Youngs modulus and R= Radius of curvature.

Fig. a

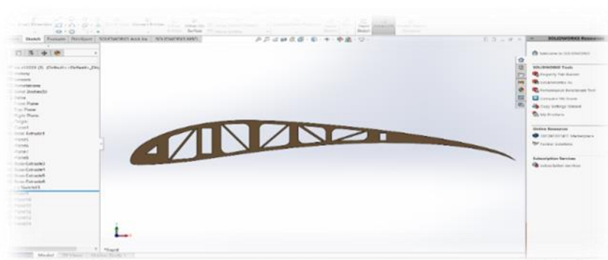


Fig. b

Factor of safety of 3 was used while substituting for 'σ' and we made sure the rib at the root bears the bending moment generated considering 'lift' as a uniformly distributed load. The maximum lift generated was calculated to be 55.42N, taking maximum co-efficient of Lift as 2.062(including High-lift devices). Surprisingly, all our designs bore this load. Intense discussions and trade-offs led us to choose design ‘Fig.a’ using Aluminium sheets and rods as spars. Three spars are used- at the location of maximum camber, 2/3rd the distance from the leading edge and in the middle in such a way that the Centre of gravity coincides with aerodynamic centre. The front and rear spars are ‘I’ section since their webs and flanges withstands axial and shear stress and a hollow-rod in the middle. Holes are made evenly in the spars to reduce weight.

The choice of distance between the ribs was an iterative process where we ran simulations by applying pressure load to the skin between two ribs and checked if it fails. Skin transmits the aerodynamic forces to the longitudinal and transverse supporting members.

We determined the magnitude of maximum Co-efficient of pressure (Cp) acting on the airfoil.

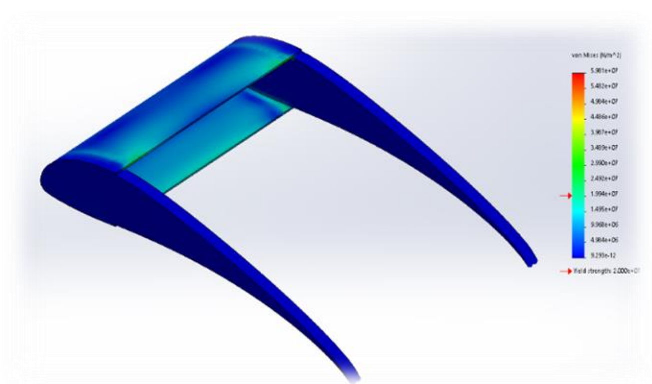
$Cp = (Po - Pa)/\ 5 * \rho * Vc^2$, from which ‘Po – Pa’ was calculated to be 350 Pascals.

Po – Stagnation pressure.

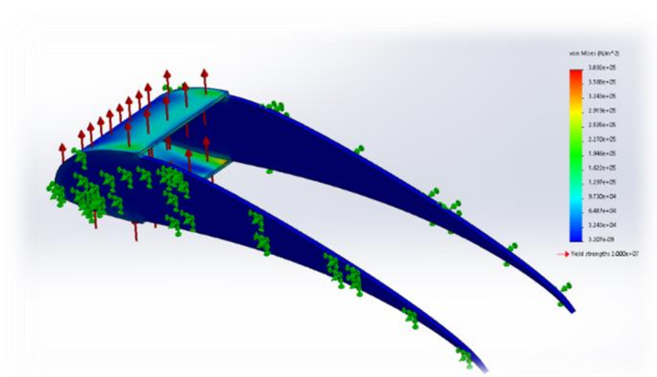
Pa- Free stream Pressure.

ρ= Density (1.225 Kg/m^3).

Vc= Cruise speed= 10 m/s.



Usually the distance between ribs were from 1-5 inches (Literature study). The above simulation performed in solid works for distance between the ribs as 3.5 inches and thickness of skin as 2mm shows that the skin buckles at a particular location, whose stress value indicated by the red arrow mark.

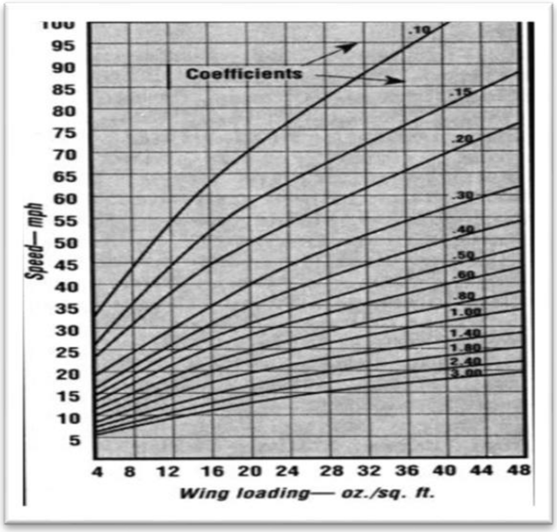


Multiple trials helped us conclude the distance between ribs as 2.8 inches and thickness of the skin used as 3mm. The material property given for ribs is that of Grain C Balsa wood and the skin is Grain A balsa wood. Here, tests were performed on skin extending only till maximum camber, after which monokote will be used as the skin. Therefore, we made sure that the entire load is borne by the skin made of balsa only.

3. ***Power Plant:***

The old saying that "you cannot get something out of nothing" is particularly true in engineering.The previous sections have discussed the aerodynamic lift and drag; the performance, stability and control of airplanes. All this takes the expenditure of *Power*, or energy which is supplied by an engine or propulsive mechanism of some type, propellers in this case. This section walks through the procedure of selection of propeller size, pitch and also about the battery selection.

Airplane wings and propeller have something in common. They are both made up of airfoil sections designed to generate an aerodynamic force. The wing force provides lift to sustain the airplane in air; the propeller force provides thrust to push the airplane forward through air. The difference is that the propeller rotates at a high RPM whereas a wing flies almost at same speed. This characteristic of propeller blade is the reason for the varying velocity from root tip; maximum at the tip and minimum in the root. The probability of propeller failing in the middle is maximum compared to other places owing to maximum stress. The stress results from the combination of centrifugal and thrust forces adding to the blade's airfoil twisting moment trying to twist them. **Diameter** and **Pitch** are the two important parameters of a propeller that decides the amount of thrust the engine produces. Diameter is the distance between tip to tip and pitch is the distance the propeller moves in one revolution if it were to move through a soft solid. Increasing both diameter and pitch would increase the load on the engine resulting in reduction of RPM. Generally, for a high-speed flight small diameter and increased pitch is used and vice versa. In the previous section, we saw that Wing Loading can vary from 16-22 oz/ft^2 from which 20 oz/ft^2 is chosen.



From graph, the required speed is found to be 40mph. This is the minimum speed that must be maintained but 25% of the speed is added resulting in 50mph owing to take-off requirements. The thrust to weight ratio (T/W) was finalised as 0.8 (previous section).

Thrust/Weight=0.8.

Thrust= 0.8 * 5700g =4600g.

Kv	Air speed (Km/h)	Air speed (mph)
1000	70	43
2000	140	87
3000	210	131
4000	280	175

Kv value	Characteristic
1000	High thrust (3.5 to 4.5 g/W approx.) Low air speed
2000	Medium thrust (2.5 to 3.5 g/W approx.) Medium to high air speed
3000	Low thrust (1.8 to 2.5 g/W approx.) high air speed (propeller and Ducted fan)
4000	Lower thrust (1.5 to 2.5 g/W approx.) Higher air speed (ducted fan)

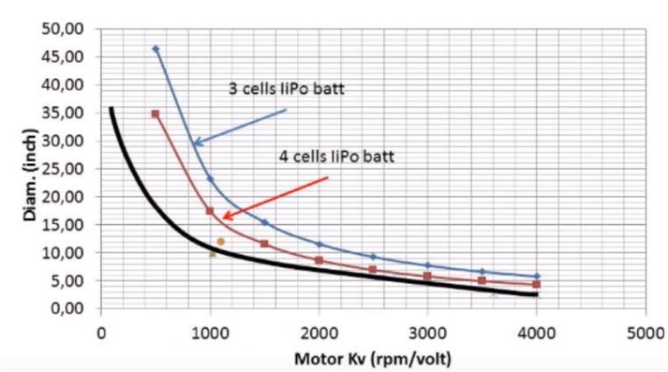
The air speed 43mph (20 m/s) is considered (little high), which will get reduced once the loads on the aircraft are considered.

A 1000 Kv motor is finalised since requirements demand only a low air speed aircraft, and as stated above, a larger propeller diameter will be used owing to the selection of relatively low Kv motor. W/g= (0.17) *(KV/1000) +0.09, the thrust to power ratio is calculated to be 0.26.

Power=0.26*4600g= 1196W.

It is always recommended to choose higher values of power (generally 40% to 50% higher) to ensure easy fly at 50 to 60% throttle. Kv= unloaded RPM/volt.

Unloaded RPM- the speed at which the propellers rotate at no load condition.



1000=unloaded RPM/14.

Unloaded RPM=14000.

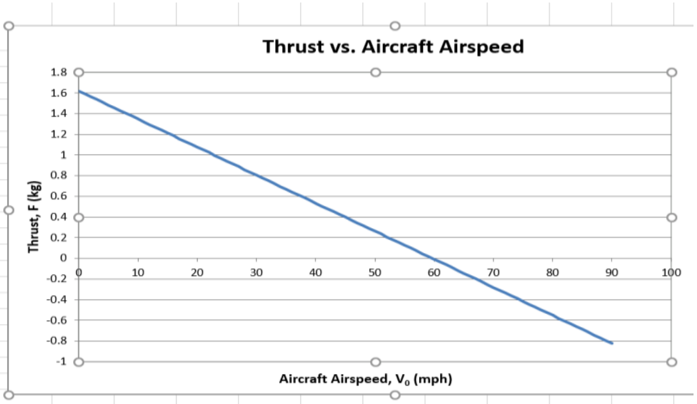
Conclusion: 12-inch propeller for 935 Kv motor and a 6-cell battery.

4.1. Static and Dynamic Thrust Calculator: The static and dynamic thrust was calculated using an online calculator. http://electricrcaircraftguy.blogspot.com/2013/09/propeller-static-dynamic-thrust-equation.html

$$F = 1.225\frac{\pi(0.0254\cdot d)^2}{4}\left[\left(RPM_{prop}\cdot 0.0254\cdot pitch\cdot\frac{1min}{60sec}\right)^2 - \left(RPM_{prop}\cdot 0.0254\cdot pitch\cdot\frac{1min}{60sec}\right)V_0\right]\left(\frac{d}{3.29546\cdot pitch}\right)^{1.5}$$

A plot in excel for various velocity starting from 0 mph till 21mph.

	Aircraft Airspeed, V_0 (m/s)	Aircraft Airspeed, V_0 (mph)	Dynamic Thrust, F (N)	Dynamic Thrust, F (g)	Dynamic Thrust, F (kg)	Dynamic Thrust, F (oz)	Dynamic Thrust, F (lb)
Static Thrust -->	0	0	15.879561	1618.712	1.618712	57.098434	3.568644016
all others are	0.44704	1	15.6133893	1591.579	1.591579	56.141355	3.508826745
dynamic thrust	0.89408	2	15.3472176	1564.446	1.564446	55.184277	3.449009474
	1.34112	3	15.0810459	1537.314	1.537314	54.227198	3.389192203
	1.78816	4	14.8148743	1510.181	1.510181	53.27012	3.329374932
	2.2352	5	14.5487026	1483.048	1.483048	52.313041	3.26955766
	2.68224	6	14.2825309	1455.915	1.455915	51.355963	3.209740389
	3.12928	7	14.0163592	1428.783	1.428783	50.398884	3.149923118
	3.57632	8	13.7501875	1401.65	1.40165	49.441806	3.090105847
	4.02336	9	13.4840158	1374.517	1.374517	48.484727	3.030288576
	4.4704	10	13.2178441	1347.385	1.347385	47.527649	2.970471305
	4.91744	11	12.9516724	1320.252	1.320252	46.57057	2.910654034
	5.36448	12	12.6855007	1293.119	1.293119	45.613492	2.850836763
	5.81152	13	12.4193291	1265.987	1.265987	44.656413	2.791019491
	6.25856	14	12.1531574	1238.854	1.238854	43.699335	2.73120222
	6.7056	15	11.8869857	1211.721	1.211721	42.742256	2.671384949
	7.15264	16	11.620814	1184.589	1.184589	41.785178	2.611567678
	7.59968	17	11.3546423	1157.456	1.157456	40.828099	2.551750407
	8.04672	18	11.0884706	1130.323	1.130323	39.871021	2.491933136
	8.49376	19	10.8222989	1103.191	1.103191	38.913942	2.432115865
	8.9408	20	10.5561272	1076.058	1.076058	37.956864	2.372298594
	9.38784	21	10.2899555	1048.925	1.048925	36.999785	2.312481322



4. ***Servo-Sizing.***

The Control surfaces of the aircraft are actuated using servos. In order to determine the apt servo size, it is necessary to calculate the torque required to move the control surface. Accordingly, the servo size is determined.

The Formula used is- **Torque (oz.-in) = 8.5E-6 * (C 2* V 2* L* sin(S1) * tan(S1) /tan(S2)).**

C = Control surface chord (cm). L = Control surface length (cm).

V = Speed (MPH). S1 = Max control surface deflection in degrees.

S2 = Max servo deflection in degrees.

	S1	S2		oz-in		
	Control Surface Deflection	Servo Max Deflection	MPH	Torque	Chord(cm)	Span(cm)
Aileron	45	50	60	16.95	6.9	48.36
Elevator	45	50	60	16.34	5.84	65
Rudder	45	45	60	7.6	5.68	26.74

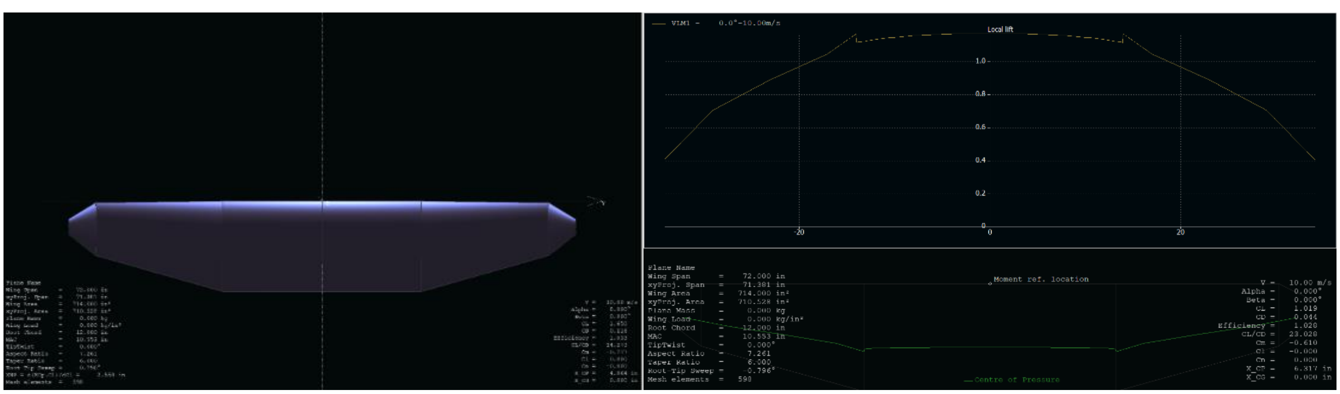
The RC plane is generally operated at a velocity of 40MPH. 60MPH is taken for safer side. The Torque requirements of the control surfaces can be achieved by EMAX ES08A Mini Plastic gear servo 8.5g, that produce 1.5 Kgf-cm torque @4.8V.

5. *Conclusion:*

We are extremely grateful for having been given the opportunity to perform this project, through which we have undoubtedly gained a plethora of knowledge. The entire designing phase was challenging and interesting. Every subsystem was preceded by multiple discussions, and sometimes, we discarded and started afresh on our work as well. Although many our results proved to be positive theoretically, following our intuition and gut feeling delayed the process of finalising sub-systems. Hence, we created and tested a scaled down prototype of the design without landing gear (to evaluate wing, horizontal and vertical stabilizer performances). Flight performances were observed, noted and altered.

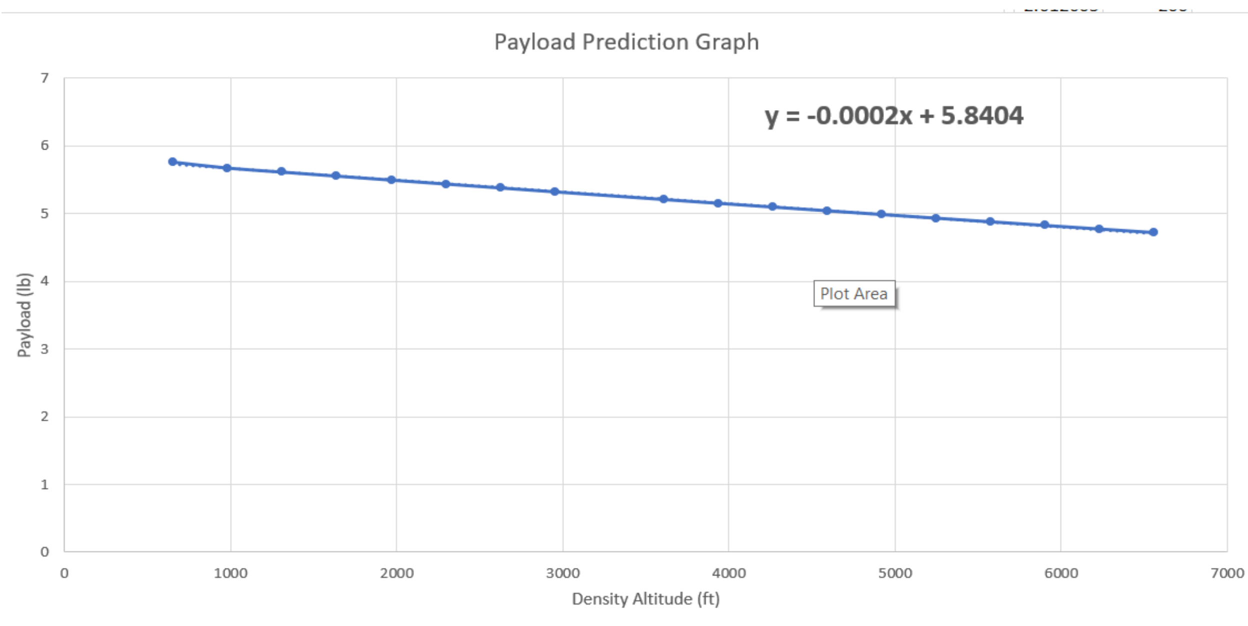
The prototype model:

Our Final Wing Design and lift distribution in XFLR file.



6. ***Payload Prediction Graph***

Payload lifted is dependent on many parameters and one such is density. A graph between payload and density altitude is plotted using Lift equation and density altitude data obtained from John. D. Anderson's 'Introduction to Flight'.



Co-ordinates sheet:

1	T	rho	w	payload	h in m	rho(kg/m3	lb	Dens in ft
2	287.4	1.21419	25.78422	2.627227	100	1.2133		
3	286.8	1.203454	25.53588	2.612603	200	1.2071	5.759797	656.168
4	286.2	1.19279	25.28923	2.572507	300	1.1901	5.671401	984.252
5	285.6	1.182198	25.04425	2.545619	400	1.1787	5.612123	1312.336
6	285	1.171679	24.80094	2.518731	500	1.1673	5.552845	1640.42
7	284.4	1.161231	24.55928	2.492079	600	1.156	5.494088	1968.504
8	283.8	1.150855	24.31928	2.465663	700	1.1448	5.43585	2296.588
9	283.2	1.14055	24.08092	2.439483	800	1.1337	5.378132	2624.672
10	282.6	1.130316	23.8442	2.413302	900	1.1226	5.320414	2952.756
11	282	1.120151	23.6091	2.400094	1000	1.117	5.207059	3608.924
12	281.4	1.110057	23.37563	2.361885	1100	1.1008	5.150901	3937.008
13	280.8	1.100033	23.14376	2.336412	1200	1.09	5.095263	4265.092
14	280.2	1.090078	22.9135	2.311175	1300	1.0793	5.040145	4593.176
15	279.6	1.080192	22.68484	2.286174	1400	1.0687	4.985027	4921.26
16	279	1.070375	22.45777	2.261173	1500	1.0581	4.930429	5249.344
17	278.4	1.060626	22.23228	2.236408	1600	1.0476	4.876871	5577.428
18	277.8	1.050945	22.00836	2.212114	1700	1.0373	4.822793	5905.512
19	277.2	1.041332	21.78601	2.187585	1800	1.0269	4.769755	6233.596
20				2.163527	1900	1.0167	4.717237	6561.68
21				2.139705	2000	1.0066		
22								

KINDLY CONSIDER THIS AS 31ST PAGE AND 2D DRAWING AS 30TH PAGE.

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