TECHNICAL DESIGN REPORT

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The following design report describes our team's design process philosophy and the reasoning behind the decisions taken w.r.t. design of a Micro Class RC aircraft under the organizers' constraints. The competition's main aim is to develop an RC aircraft to carry and deliver PPE kits and sanitizer bottle and drop two payloads at different locations. The design objective is to create an RC aeroplane capable of producing high lift and carrying as much payload as possible and a stable flight. The RC aeroplane is a fixed-wing aircraft, but the wings, stabilizers many other can be detached from it, which makes it packing efficient. The aircraft should also move on the runway without the use of control surfaces.

1. Airfoil Design

We initially considered airfoils that will have favourable characteristics in the conditions at which our plane will fly. Using this, we narrowed the number of airfoils down to 6, namely GOE 525, S 1223, S 1223RTL, S 1210, E 420 and E 423. An analysis was carried out for these six airfoils in XFLR. The S1223 airfoil had the best lift characteristic and had a decent drag characteristic too. But for Reynolds number values in the range 50k to 100k, it had a sharp drop in C_L value of about 0.5 at a low angle of attacks. For Reynolds number above 100k, the dip does not occur, and there is a considerable improvement in the lift characteristic. The S1210 airfoil had the second-best lift characteristic. It did not show any noticeable dip as seen for the S1223 airfoil. So, we decided to have a blend of both the airfoils by interpolating the coordinates of “S1223” and “S1210”, gives different weightage to both the airfoils.

The sharp dip in the C_L value was found to disappear only for less than 30% weightage to S1223. But more the S1210 character, lesser was the lift produced by the airfoil. **So, 70% S1210 - 30% S1223 blend was concluded to be optimum and was chosen**. By interpolating, the C_L dip disappeared, and the drag character improved at the cost of a small decrease in lift produced.

Figure 1 shows the characteristics of the chosen airfoil. The crowded curves are the ones which fall in the operating Reynolds number range.

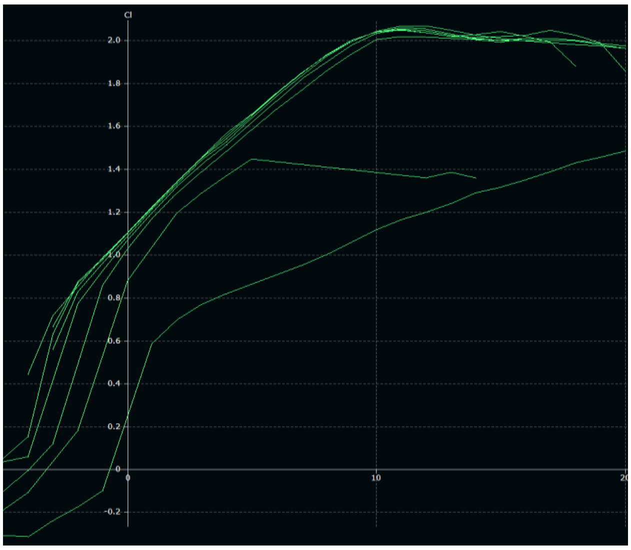


Fig. 1

2. Wing Planform Considerations

With the selected airfoil, the planform for the wing must be considered. The primary requirement of the wing planform is to be able to produce enough lift to handle the dry weight and the payload. Though the elliptical planform is very efficient, we have not considered it, because of its manufacturing complexity. The backward tapered and trapezoidal planforms were not chosen for iteration because of the problem of limited roll control of the aircraft as the ailerons would be at an angle to the relative airflow allowing us to use only the perpendicular component of the velocity of the wind which decreases the aileron efficiency.

We Iterated the rectangular and forward tapered planforms in XFLR5 for different aspect ratios (around 5-7) to find dimensions for root chord and tip chord for the two planforms that would produce the optimum lift while still maintaining manoeuvrability. The following values provided the most desirable characteristics for each planform considered at 10 m/s free stream velocity.

Planform	Span(m)	Root chord(m)	Tip Chord(m)	Cl at 0° angle of incidence	Cd at 0° angle of incidence	Cl at 3° angle of incidence	Cd at 3° angle of incidence	The weight lifted at a 0° angle of incidence (kg)	The weight lifted at a 3° angle of incidence (kg)
Rectangular	1.35	0.27	0.27	0.786	0.04	0.99	0.06	1.757	2.213
Forward tapered	1.35	0.35	0.21	0.793	0.04	1.001	0.06	1.836	2.313

Based on these values, considering the highest sample lift produced by the backward tapered wing planform (as measured through XFLR5), lower wingtip vortices and subsequent vortex drag and the roll control advantages, we chose the forward tapered wing **with a 0° angle of incidence** (To prevent stall during takeoff while still maintaining sufficient lift) **1.35m span, 0.35m root chord and 0.21m tip chord** to further analyze in ANSYS to get more accurate values of lift production. We analyzed this airfoil in ANSYS for values of free stream velocity from 5-16 m/s for lift and drag force.

Velocity (m/s)	Lift (N)	Weight (kg)	Drag(N)
8	11.773	1.200	1.183
9	14.960	1.525	1.485
10	18.559	1.896	1.833
11	22.464	2.290	2.215
12	26.771	2.729	2.629
13	31.412	3.202	3.071
14	36.717	3.743	3.549

We decided to go with a **high wing configuration** because of high ground clearance, **shorter landing distance** because of lesser impact of ground effect, **inherently stable characteristics since the centre of mass is situated below the centre of lift.**

3. Velocity Estimate, Motor and propeller combination

Estimation of the velocity is the primary task as it helps to decide on the Reynolds number to aid further analysis and to determine the motor and propeller configuration.

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

Where density = 1.225 kg/m^3 and viscosity = 1.81 X 10^{-5} kg/m.s.

The equation reduces to Re = 67679.59.56*v*c.

To determine the velocity of the plane, we plotted the thrust and drag of the plane against velocity. The values of drag were calculated using drag analysis in ANSYS. The Thrust equation is

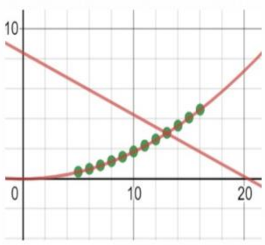
$$T(v, \omega) = \tfrac{1}{2}\rho A(0.7R\omega)(\sqrt{(0.7R\omega)^2 + v^2})\left(C_{L_o} + C_{L_\alpha}(\theta_p - (\tfrac{v}{0.7R\omega}))\right)$$

On plotting the above equation, we noticed that the variation of Dynamic Thrust with the velocity of plane varied linearly. Hence, we reduced the equation to a simpler form, given by the static thrust and the maximum velocity, which was determined using the propeller pitch and given RPM from the datasheet.

$$\frac{T}{To} + \frac{v}{vo} = 1$$

We used AT2317 1250KV motor powered by LIPO 4S-2200 mAh battery. Using the thrust data, a thrust vs velocity graph was also plotted on the same graph. The point of intersection is the predicted velocity which is equal to 12.9 m/s at which the aircraft should cruise—the pitch and the diameter of the propeller 6 and 9 inches, respectively.

Propeller	Throttle	Voltage (V)	Current (A)	Power (W)	RPM	Torque (N*m)	Thrust (g)	Efficiency (g/W)
APC 9*6	40%	15.09	9.06	136.63	7581	0.114	749	5.48
	45%	15.06	10.66	160.19	8013	0.128	840	5.24
	50%	15.02	12.37	185.59	8418	0.142	937	5.05
	55%	14.97	14.16	212.01	8774	0.156	1050	4.95
	60%	14.93	16.10	240.25	9130	0.171	1131	4.71
	65%	14.88	17.91	266.47	9474	0.185	1217	4.57
	70%	14.82	20.45	302.98	9880	0.203	1341	4.42
	75%	14.72	23.84	350.85	10287	0.227	1457	4.15
	80%	14.61	27.14	396.58	10619	0.249	1562	3.94
	90%	14.32	34.69	496.77	11369	0.288	1696	3.41
	100%	14.23	36.73	522.62	11500	0.295	1733	3.32



speed(m/s)	Drag (N)
7	1.2171
8	1.5867
9	2.0051
10	2.4719
11	2.9873
12	3.5518
13	4.1646
14	4.8275
15	5.5379
16	6.2953
17	7.1034
18	7.9573

Fig. 2 Fig. 3 Fig. 4

4. Servo sizing

The proper servo must be selected to ensure the functionality of all control surfaces. Each surface experiences a different amount of force, so each servo needs to be sized according to its respective control surface. Torque calculations are dependent on the control surface chord C, velocity V, control surface length L, maximum control surface deflectionS1, and max servo deflection S2. The below Equation outlines the appropriate torque calculations for the servo sizing.

$$T\ (oz-in) = 8.56 \times 10^{-6}\left(\frac{C^2V^2L\sin(S_1)}{\tan\,(S_2)}\right)$$

Control Surface Max deflection from center (degrees) = 80, Servo max deflection from center (degrees) = 30, Velocity = 560.

The above equation generates torque values for the ailerons, rudder, and elevator as 8.82744361 oz-in, 16.04477952 oz-in, 9.223777812 oz-in respectively, and these torque values are under the limits of the servo.

5. Takeoff Velocity, Takeoff Distance and Takeoff performance

Takeoff Velocity was calculated equating Lift equation = weight of the aircraft (with payload) and was found to be 12 m/s. A drag function for the plane and the thrust function for the propeller was constructed.

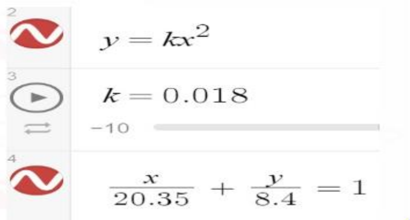


Fig. 5

Equation 4 represents thrust, and equation 2 represents drag, now we know thrust – drag= mass * acceleration, we found out the acceleration. From that, we were able to find the distance as a function of velocity. The takeoff distance was found out to be 19.7 m. We also found out that the max angle with the ground will be 28 degrees for takeoff from the equation Sin B = (T-D)/W. Velocity of stall during takeoff will be equal to V_{to} / 1.2 = 10 m/s.

6. Landing performance and impact analysis

The impact loading on the plane's body is calculated from the impulse generated during the touch down after the flare-up. With a $C_{L_{max}}$ Value of **2.063** calculated from the graph plotted using Reynold's number during cruise flight, the velocity of stall of the empty plane is calculated to be **4.68 m/s**.

$$V_{stall} = \sqrt{\frac{2 \times w/s}{\rho \times C_{L_{max}}}}$$

The value of the touchdown velocity is calculated to be 5.39m/s using the formulae

$$V_{TD} = 1.15 \times V_{stall}$$

Assuming a flight path angle of **-5°**, the vertical component of the velocity at touch down is **0.469 m/s**. The momentum experienced by the empty plane (weighing around 1kg) at touch down is **0.938 kg.m/s**. Assuming, a flare up time of 1 second, the force experienced is **0.938 N.**

In order to account for the sudden impact load generated on landing, the fuselage is made from Corrugated Polypropylene Copolymer fluted sheets which can handle the impact as explained later.

7. Structure and Structural Analysis

Wing Structure:

Considering the airfoil considered and its curvature, two materials of wings were considered. A high-density foam wing of complete planform volume and an aeroply based airfoil structure where leaf-like airfoil of 1.5 to 3mm thickness was made at certain intervals to make a solid wing. Based on weight considerations, the airfoiled aeroply wing was selected rather than full volume solid foam wing in order to account for the optimum weight of the plane with payload. The net stable wing structure weight is shown in the table below:

Part	Foam Wing(g)	Airfoil Wing(g)
Solid Structural Element	283	156.83
Carbon Rod	70	63.94
Airfoil Structure (External in foam, Internal in aeroply)	160	169.95

Based on the net weight of the overall structure, aeroply based airfoil wing was selected.

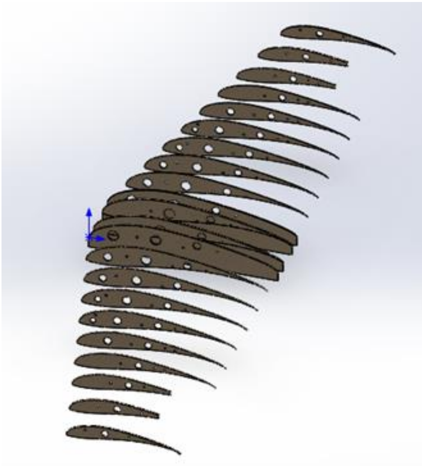


Fig. 6

As can be seen, airfoil profiles were made by scaling the airfoil coordinates from a chord length of 35cm to 21 cm with regular intervals of 7cm. These airfoils are covered with a thin sheet of monocot plastic which ensures that the required aerodynamic shape is maintained. These are further supported by 3mm and 6mm carbon fibre rods running either entirely through the wing or through areas of concentrated pressure forces.

One 704 mm, 1480mm, 1480mm and two 365 mm carbon fibre rods are kept at an appropriate spacing from the leading edge of the central airfoil profile. The structure was analyzed for stress considerations and deflection under maximum load under considerations and the highest lift possible at the cruise speed of 12m/s. Using these several iterations were made to make slight adjustments to stress and strain consideration to ensure the least deflection and minimum stress. Von-mises analysis was carried out for the so formed airfoil-based wing structure with a net lift of 2.15 kgf and a weight of 1.8 kgf acting on the wing. The values so obtained are indicated in the table below.

Name	**Type**	**Min**	**Max**

Static Stress	Von Mises	7.51685e-007 N/m^2	1676.49 N/m^2

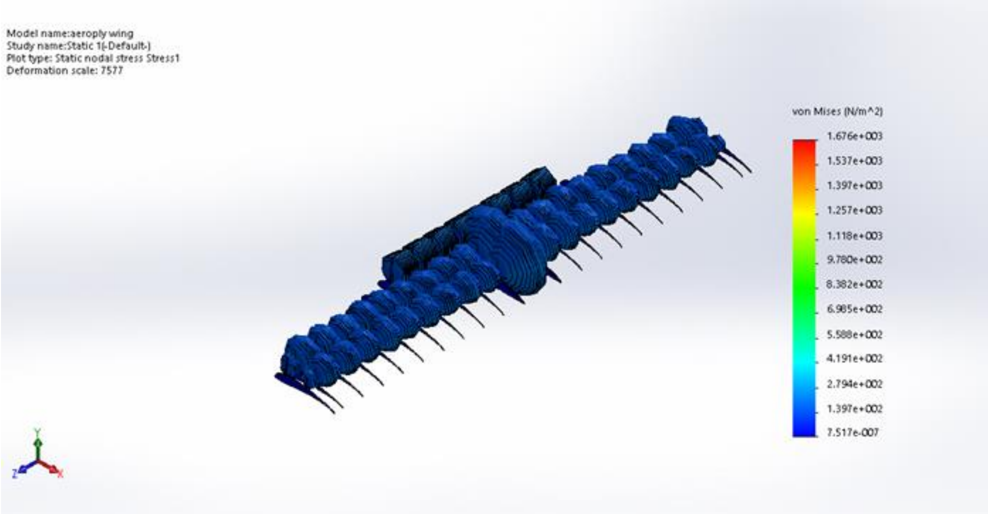
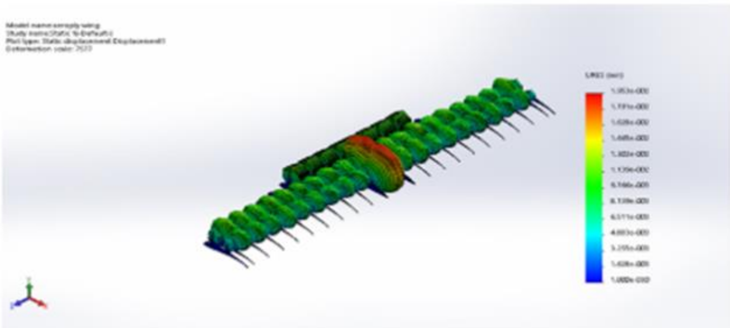


Fig. 7: Aeroply Wing Stress Analysis

The stress remains nearly constant throughout the structure for the currently bonded wing stricture and is close to the minimum value of stress throughout all rods. The only high-stress point exists at the upper planes of the centre airfoils which is taken care of by the static strength of the fuselage and a second set of inner airfoils at the centre.

Name	Type	Min	Max
Displacement	URES: Resultant Displacement	0 mm Node: 21755	0.0195328 mm Node: 14511



Aeroply Wing Displacement Analysis

Fig. 8

As can be seen from the displacement plot, the rods undergo a net positive bending moment (sagging) such that the maximum vertical displacement occurs in the backward rods. However, these deflections are negligible and accounted for by the cover binding of the wing.

Aileron Structure:

The aileron dimensions designed based on analysis (through iterations in ANSYS). An aileron of 23 cm x 6cm of a rectangular planform with a rib structure like that of the wing. It is an internal structure made through airfoil cuts in the outer airfoils on both sides (Fig. 9). The overall structure is made put of aeroply.

The airfoil is connected on two sides through 2 x 255 mm carbon fibre rods which hold the aileron system in position. The carbon fibre rods are glued to the holes in the surrounding ailerons and is lubricated around the holes attached to the ailerons. The ailerons are connected to a horn which further connects it to a servo for movement.

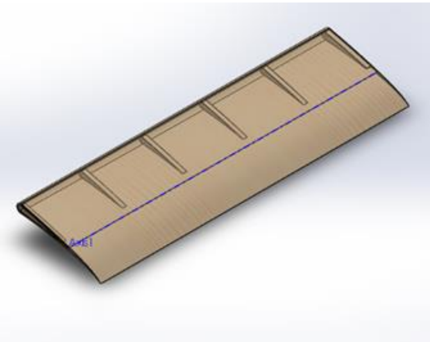
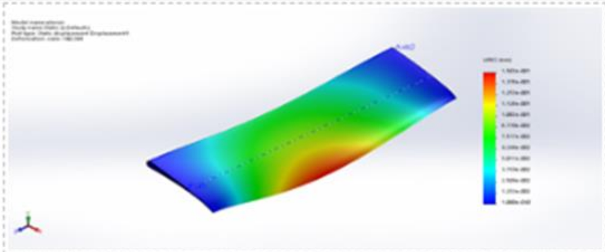


Fig. 9

Name	Type	Min	Max
Displacement Test	URES: Resultant Displacement	0 mm	0.150343 mm



Aileron Displacement Analysis

Fig. 10

The structure was analyzed for the harshest condition possible for the given aileron, i.e., due to strong upper wind flow on the action of downward weight and the maximum deflection was notes as follows. The deflections were minimized by using airfoil-shaped mid trusses which prevent mutual deflection to a large extent.

Overall Wing Structure

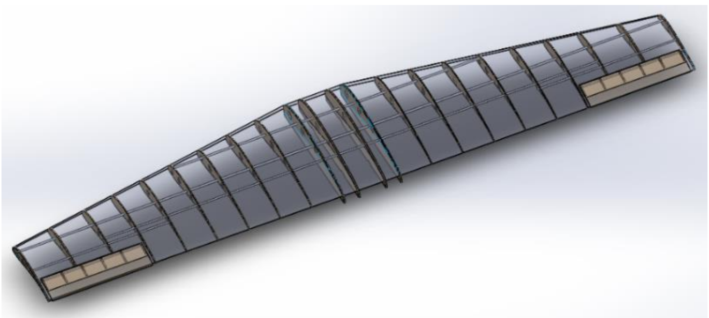


Fig. 11

Fuselage and Strength Control

In order to reduce the drag of the fuselage, a shape inspired by airfoils was chosen. The plane's plan and side profiles are based on the <plan airfoil> and <side airfoil>, respectively.

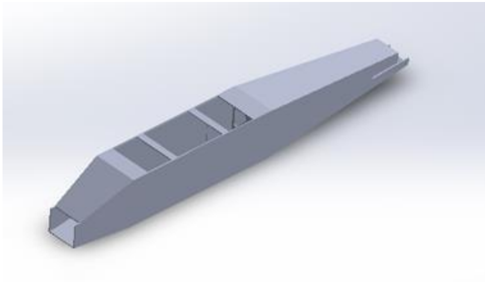


Fig. 12

As observed from the side, the fuselage has a gentle taper towards the back and a slightly sharper taper at the rear. It allows the flow over the plane, to stick to the surface and reduces any excess drag produced. Moreover, if we observe the top view, the flight again follows a gentle taper at the front and the back, this time, symmetric across both sides.

Furthermore, U shaped walls are attached to the main coroplast wall for structural rigidity against loads on the side. Cross-sectional trusses have been provided throughout the fuselage to aid in this regard.

As mentioned earlier, the fuselage has been designed considering takeoff and landing into account in terms of the thrust and reaction. As mentioned, the maximum impact load on landing is 0.938N applied through the time of impact/deformation. After several iterations, 3mm thickness Coroplast Sheet fluted with 0.008mm thickness sheets are used.

Name	**Type**	**Min**	**Max**
Deflection Test	URES: Resultant Displacement	0 mm	0.000676492 mm (Red Region)

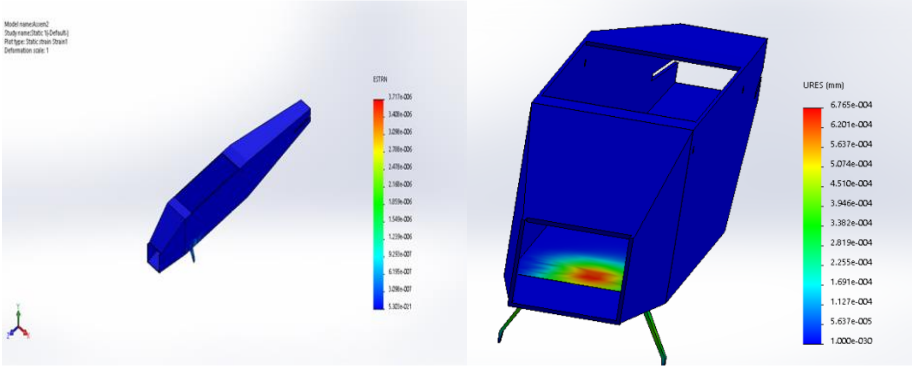


Fig. 13: Concentrated Impact Test

The Coroplast sheet, through its **I**-section system, does not allow a massive impact on the fuselage. Similarly, this section accounts for the wind thrust and the boundary layer displacement forces on the sidewalls. However, they produce negligible deflection.

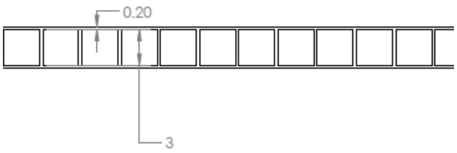


Fig. 14: Coroplast 3mm thick

8. Stability and Control

Empennage Design

The tail design is done by using the tail volume coefficient method to estimate the required empennage dimensions. The tail volume coefficients for horizontal and vertical tail are given by

$$C_{Ht} = \frac{L_{Ht} \times S_{Ht}}{C_w \times S_W}$$

$$C_{Vt} = \frac{L_{Vt} \times S_{Vt}}{C_w \times S_W}$$

L_{Ht} and L_{Vt} are called moment arms which are commonly approximated as the distance between 25% of wing mean chord length to the tail quarter chord length of the vertical and horizontal stabilizer respectively. Cw is wingspan while S denotes the surface area.

Typical values for C_{Ht} are 0.2 to 0.7 and for C_{Vt} are 0.02 to 0.07. The higher the number, the more stability achieved. We initially started with C_{Ht} of 0.5, and then multiple iterations were performed in order to achieve positive static stability which is explained in detail under static stability. We finally got with C_{Ht} of 0.64. Also, for vertical stabilizer design, appropriate crosswind analysis was performed in ANSYS, which is elaborated under Rudder and VS design. We achieved C_{Vt} of 0.02, which is in the range of typical values. **By rearranging the equation and selecting a suitable tail volume coefficient, required empennage area can be calculated and then suitable dimensions for HS and VS were determined.**

Longitudinal Static Stability Analysis

The stability of the plane along the longitudinal axis varies according to the centre of gravity and neutral point. Generally, COG does not coincide with the point at resultant lift works, so a pitching moment is produced, making the plane unstable. The horizontal stabilizer produces counter torque to stabilize aircraft, so designing the horizontal stabilizer is very crucial. A general moment equation was generated to analyze the plane's stability about the longitudinal axis and parameters affecting it, which is given below, and all the data was analyzed using xflr5.

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

C_m is moment coefficient, C_{mac} is the moment of the wing about the Mean Aerodynamic Centre, C_L is the lift coefficient of plane and C_{LE} is the lift coefficient of the horizontal stabilizer, and V_H is volume coefficient of horizontal stabilizer. Multiple iterations were performed according to tail volume coefficient to achieve positive static stability.

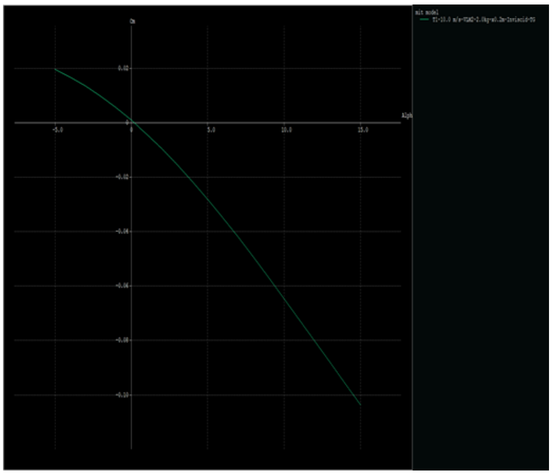


Fig. 15

This is the C_m versus alpha graph. Negative slopes ensure a longitudinally stable flight. At the operating angle for cruising, which is nearly zero degree, the C_m.is almost zero which ensures the equilibrium during cruise. **The position of neutral point is around 21 cm from leading edge of wing**. Now the degree of stability depends on static margin which is given by

$$S.M. = \frac{X_{np} - X_{cg}}{c}$$

Knowing the position of CG, which is 18.6 cm from the leading edge of the wing, we can calculate the value of the static margin coming around 0.0827. Generally, for most stable planes, static margin values are between 0.05 to 0.25, with stability getting smaller with lesser values of static margin. In our case, the static margin value is 0.08, which is quite good because the value falls in mid-range, so the plane is not too unstable or too stable to resist the maneuvers. Also, the C_m value at our operating angle of attack (0 degrees) is approximately zero which ensures equilibrium during cruise stage.

<u>Control Surface Sizing:</u>

<u>(i) Aileron sizing</u>

Design outcome: aileron activation must cause the plane to rotate 1 radian in 1 second

Constraints/Assumptions: Maximum angle of aileron deflection is 30 degrees, the ratio of aileron width to MAC of the plane is 0.25, the aileron will be located at the tip of the span to maximize moment.

Procedure: An Analytic equation was developed that described the angle rotated as a function of time and aileron deflection. This equation is valid under the assumption that the roll resistance of the aircraft is directly proportional to the angular velocity which holds when velocity is much higher compared to product of angular velocity and half-span.

$$\theta = \frac{K_{ail}}{K_{gli}} t - \frac{K_{ail}}{K_{gli}^2}\left(1 - e^{-\frac{K_{pla}}{I}t}\right) \quad (1)$$

K_{ail} – Aileron dependent constant, which is derived from the relation given below, from the paper Programmable Aileron Sizing Algorithm for use in preliminary aircraft design software, published in the journal of engineering and applied sciences.

$$C_{l_{\delta_a}} = \frac{C_{L_{\alpha_w}} \tau_a C_r}{Sb}\left[\frac{y^2}{2} + \frac{2}{3}\left(\frac{\lambda-1}{b}\right)y^3\right]_{y_i}^{y_o} \quad (2)$$

K_{gli} – Glider dependent constant, is the proportionality constant between the roll resistance of the glider and the angular velocity. Computed by hand assuming a radial variation of the angle of attack across all surfaces and an extra component was added for the fuselage.

I – Moment of Inertia of the plane about the roll axis. Computed using "Solidworks".

By substituting required values in (1) we can solve for required value of K_{ail} and using (2) we can find the inboard location of the aileron assuming tip of span to be outboard position hence solving for aileron size.

All required constants were either present in the paper or were computed from Xflr5.

The final dimensions of ailerons: chord = 6 cm, span = 20 cm, ailerons are located at the tip of the wing.

(ii) Rudder/VS sizing

Design outcome: the plane must be able to withstand a crosswind of 4m/s

Assumptions/Constraints: Rudder chord is assumed to be around 0.6 of the VS to ensure enough extra length at the tip chord for structural soundness. The maximum deflection of rudder was 45 degrees.

Procedure: Analytic relations describing Rudder behaviour was taken from the paper: An Educational Rudder Sizing Algorithm for Utilization in Aircraft Design Software published in the International Journal of Applied Engineering Research.

$$C_{n_{\delta r}} = -C_{l_{\alpha_v}} \forall_v \eta_v \tau_r \frac{b_r}{b_v} \qquad (1)$$

$$N_a = qSbC_n \qquad (2)$$

Using the above equations, we can determine the rudder's functions as a function of the rudder's properties and the vertical stabilizer. To compute the wind's effects, an ANSYS analysis is made where there is a crosswind of 4m/s and 12m/s headwind when the glider is in steady flight condition on a model of the glider without the rudder. The yaw moment due to crosswind is Na 0.65 Nm. Using the side drag, we can calculate a VS-rudder combo that can keep the plane stable under such circumstances by analyzing both the effects of the rudder and the influence of crosswind on the VS, causing it to create a yawing moment.

The final dimensions of VS/ Rudder:

VS root chord: 11 cm, VS tip chord: 10 cm, VS span: 14 cm

Rudder chord: 5 cm, Rudder span: 14 cm, Rudder area: 0.007 m^2

(iii) Elevator sizing

Design outcome: Elevator activation must cause the plane to rotate 1 radian in 1 second

Constrains/assumptions: span of the elevator is the same as the horizontal stabilizer, maximum deflection of the elevator is 30 deg.

Procedure: An analytical relation describing Elevator behaviour was taken, which is given below:

$$C_{L_{\delta_e}} = C_{L_{\alpha_h}} \eta_h \frac{S_h}{S} \frac{b_e}{b_h} \tau_e$$

Using the above equation, we can determine the elevator's properties as a function of the elevator's properties and the horizontal stabilizer. The pitching motion will change the lift and, consequently, the moment, so taking the moment of it about cog and simplifying the relation can get the above equation. Plugging in required values in the equation, we can get the elevator area from which chord length can be decided, which we got 40% of HS's chord.

The final dimensions of elevator: Span = 52 cm, chord: 8 cm

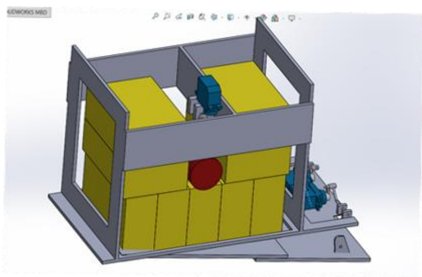
9. Electronics Selection

- Battery – 4S 2200 mAh
- Motor – AT237 Long Shaft 1250Kv
- ESC – 40A
- Servo motors – SG90 1.8 kgf*cm

10. Scoring analysis of payload:

FS=50*(No. Of PPE Containers (Cuboid)) + 20*(No. Of hand sanitizer units (Cylinder))

Since the net weight of the empty plane is around 1kg, and the net lift produced is 2.2 kg, this allows a maximum payload of 1kg with a factor of safety. Considering this to be a linearly programmed problem, we get the following conditions:



For A and B being the number of cuboids and cylinders respectively. Maximize F=50A+20B (Flight Score)

Fig. 16

Subject to: A>=1 (At least one cuboid); B>=1 (At least one cylinder) 100A+50B<=1000 (Net weight less than 1kg=1000g). We need to take in the maximum number of cuboids along with the above-mentioned criteria order to get the highest possible score. The constraints were plotted, and the feasible region was noted. Based on these two combinations of cylinders and cuboids were considered within the feasible region, as mentioned below. Of these, **the combination of nine cuboids and two cylinders** provided the highest possible packing fraction within the dimensions of the fuselage and ensured higher points and was thus chosen.

Payload dimensions			Quantity		Weights			Scores		
length	breadth	height	cuboids	cylinders	Wcu	Wcy	total weight	Scu	Scy	total score
15	10	9	7	2	700	100	800	350	40	390
15	10	12	9	2	900	100	1000	450	40	490

Fig. 17

11. Taxi Mechanism through Steerable Tail Wheel

The tail landing gear (taildragger) is, as shown in the picture. It will be fixed to a supporting part which can be settled directly to the fuselage still the wheel can steer by the servo linkage which can be connected as a common for rudder and the steerable landing gear.

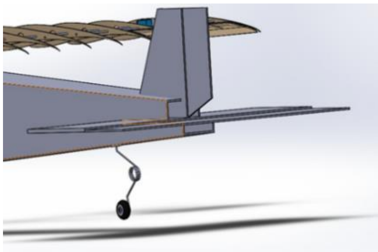
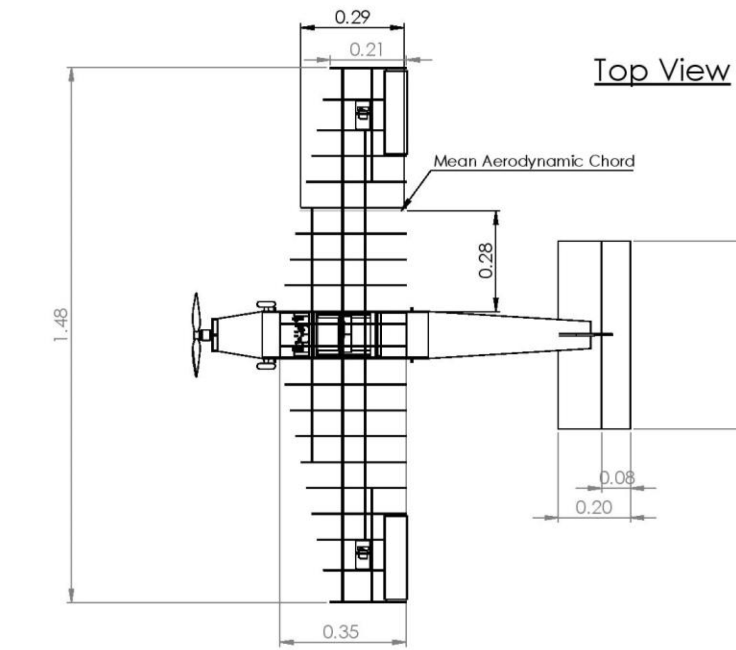


Fig. 18

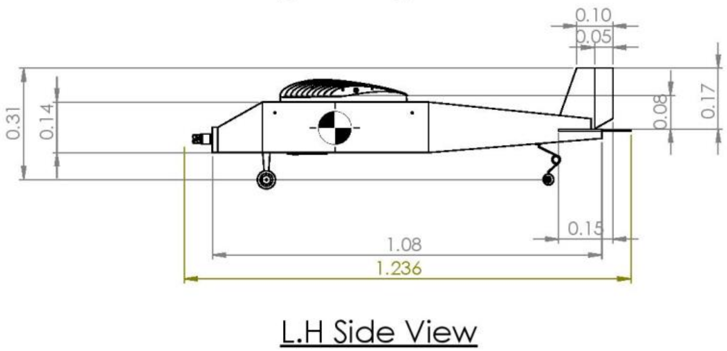
12. Table of References

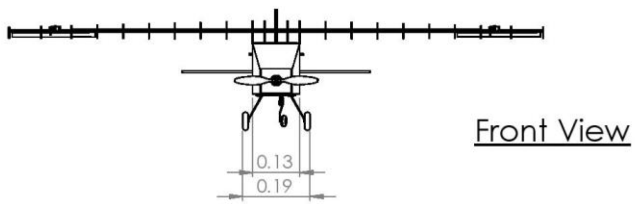
1. Aero design stuff, "The wright stuff", Department of Mechanical engineering, Northern Arizona University
2.An Aerodynamic Comparative Analysis of Airfoils for Low-speed Aircrafts, Sumit Sharma, International Journal of Engineering and Technical research
3. Programmable Aileron Sizing Algorithm for use in preliminary aircraft design software, The journal of engineering and applied sciences
4. An Educational Rudder Sizing Algorithm for Utilization in Aircraft Design Software, International Journal of Applied Engineering Research

13. 2D Drawing



Summary Table	
Empty Weight	0.994kg
Battery Capacity	2200mAh
Motor Manufacturer	T Motors
Motor Model	AS2317 Long Shaft
Motor KV Rating	1250KV
Propelller Manufacturer	Aerostar
Propeller Diameter	9''
Propeller Pitch	6''
Servo Manufacturer	Tower Pro
Servo Model	SG 90
Servo Torque	1.8 kgf*cm
ESC Manufacturer	Eflite
ESC Rating	40A
Reciever Manufacturer	Flysky
Reciever Channels	6
Wing Span	1.48m
Mean Aerodynamic Chord	0.285m
Empty C.G	0.1856m from Leading Edge of root
Fully Loaded C.G	0.1856m from Leading edge of root
Empty Stability Margin Percentage	8.27%
Fully Loaded Stability Margin Percentage	8.27%





Team Name	The Third Dimension
Team Number	5852
College	NIT Trichy
Scale of Drawing	1:11
All Dimensions in SI Units	

14. Technical Data Sheet

Team Name: The Third Dimension

Team ID: 5852

College Name: NIT Trichy

Part Name	Weight (g)	Location from Leading Edge (m)	Moment
Motor	70	-0.215	-15.05
Landing Gear Front	30	-0.047	-1.41
Landing Gear Tail	25	0.739	18.475
ESC	23	-0.096	-2.208
Propeller	15	-0.231	-3.465
Servo for Ailerons	24	0.223	5.352
Servo for Tail	36	0.779	28.044
Servo payload	24	0.072	1.728
Battery	230	-0.09	-20.7
Receiver	12	-0.12016	-1.44192
Wing	247	0.178	43.966
Fuselage truss	56	0.411	23.016
Fuselage (Coroplast)	172	0.5	86
Miscellaneous	30	0.6	18
Total	994		

Payload capacity details:

No. of cuboids = 9

No. of cylinders = 2

Total weight of cuboids carried= 9*100g = 900g

Total weight of cylinders carried= 2*50g = 100g

Total weight of the payload= 1000g

The score for cuboids = 9*50 =450 points

The score for cylinders = 2*20 =40 points

Total points = 490 points.