SAE Aero Design Challenge 2020

Design Report—Micro Class

Team Pegasus

2020 SAEISS AERO DESIGN CHALLENGE

STATEMENT OF COMPLIANCE

CERTIFICATION OF QUALIFICATION

Team Name	Team Pegasus
Team ID	ADC2020113
College/University	SASTRA Deemed to be University
Faculty Advisor	Mr. MANO. S
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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 10/1/2020

Signature of faculty advisor

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Note: A signed scanned copy of the statement should be included in your Design Report Page 2

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1. Abstract:

Our team was formed chiefly for participating in the annual competition conducted by the Society of Automotive Engineers (SAE)—The Aero Design Challenge 2020. The goal of the team is to design a Micro class UAV style aircraft that has maximum possible payload fraction, meeting all the mission requirements. The procedure significantly involved theoretical and experimental calculations. XFLR5, SOLIDWORKS and ANSYS were used in designing our aircrafts. This document describes the design of our Micro class RC plane.

2. Introduction:

Society of Automotive Engineers (SAE) conducts the Aero Design Challenge annually providing undergraduate and graduate engineering students a real-life engineering challenge. Unmanned Aerial Vehicles are used in many civil applications owing to their ease of deployment, low maintenance cost and high mobility.

3. Requirements:

Table 1: Mission Requirements

Weight	Empty weight of the plane should not exceed 1.5 kg
Dimensions	Dismantled plane should be accommodated in a 3 feet cubic box
Payload	Must weigh as much as possible but payload bay be of dimensions 5” x 1.5” x 1.5”
Flight path	The plane must be hand launched, travel in the stipulated path and land in the landing zone

4. Design Process:

Aircraft designing is an iterative process. It is a compromise between various competing factors and constraints to produce best aircraft. Designing could be said as a process of devising a system, component or process to meet desired needs. This involves application of basic sciences, mathematics and engineering sciences optimally to convert resources to meet the stated objective.

The root of our aircraft's specifications lies in exploring many other RC planes, gliders and cargo planes. This helped in developing the conceptual design. Also, the data helped us to assess the advantages and disadvantages of various configurations of the airplane parts. Simulations were done to adopt best airfoil for the wing, best wing configuration to attain higher lift and easy maneuverability.

4.1. Main Wing:

The main wing is the most crucial part of any airplane because it is responsible for production of most of the necessary lift. Most of the other parts of the aircraft depend on the configuration of the main wing (like area of horizontal and vertical stabilizers).

The objectives like maximum lift, minimum drag, ease of maneuverability was taken into consideration during the design of the main wing.

4.1.1. Airfoil Selection:

The cut cross section of the main wing is said to be its airfoil. There are many kinds of airfoil shapes each having their own pros and cons. Selection of the airfoil shape was the first step in designing the wing. The following parameters were predominantly investigated.

i) Coefficient of lift for varying Angles of Attack (C_l vs. AOA)

ii) Coefficient of lift for varying Coefficient of Drag (C_l vs. C_d)

iii) Ratio of Coefficients of lift and drag for varying Angles of Attack (C_l/C_d vs. AOA)

iv) Coefficient of Drag for varying Angles of Attack (C_d vs. AOA)

These attributes were examined upon Reynolds numbers 100000-200000. The Reynolds number was calculated for the assumed aircraft speed of 10-12 m/s, chord length of 20 cm and environmental conditions. After a detailed survey, our list of airfoils reduced to three S1223, E193 and NACA 2410. The characteristics of these airfoils are discussed below:

- **S1223:**

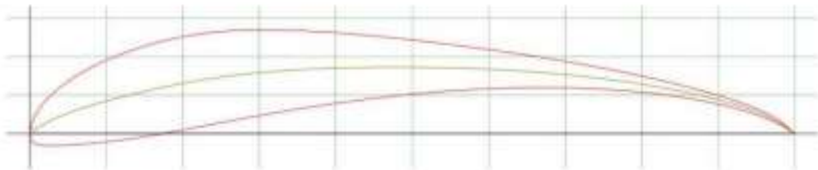


Figure 1: S1223 airfoil shape

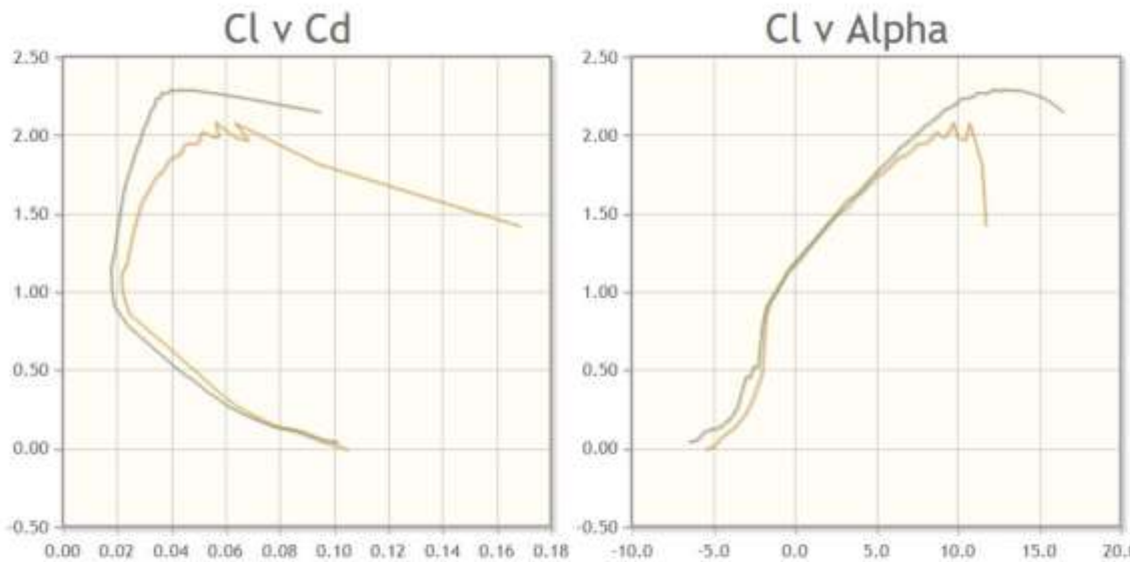


Figure 2: C_L vs C_D and C_L vs Alpha for S1223 airfoil

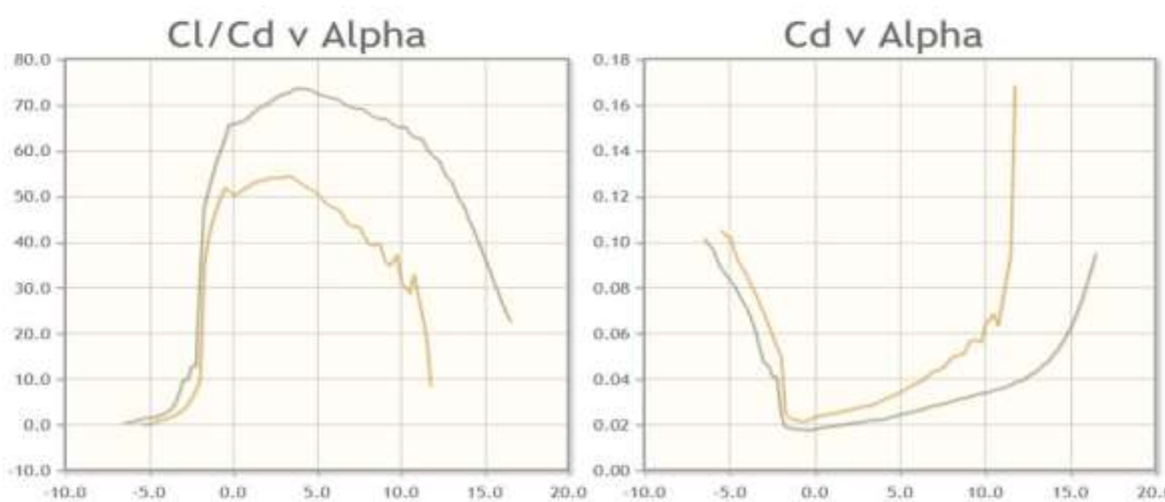


Figure 3: C_L/C_D vs Alpha and C_D vs Alpha for S1223 airfoil

- **E193:**

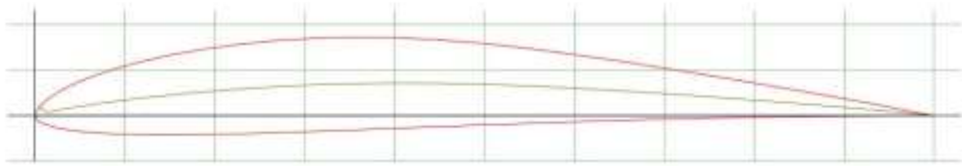


Figure 4: E193 airfoil shape

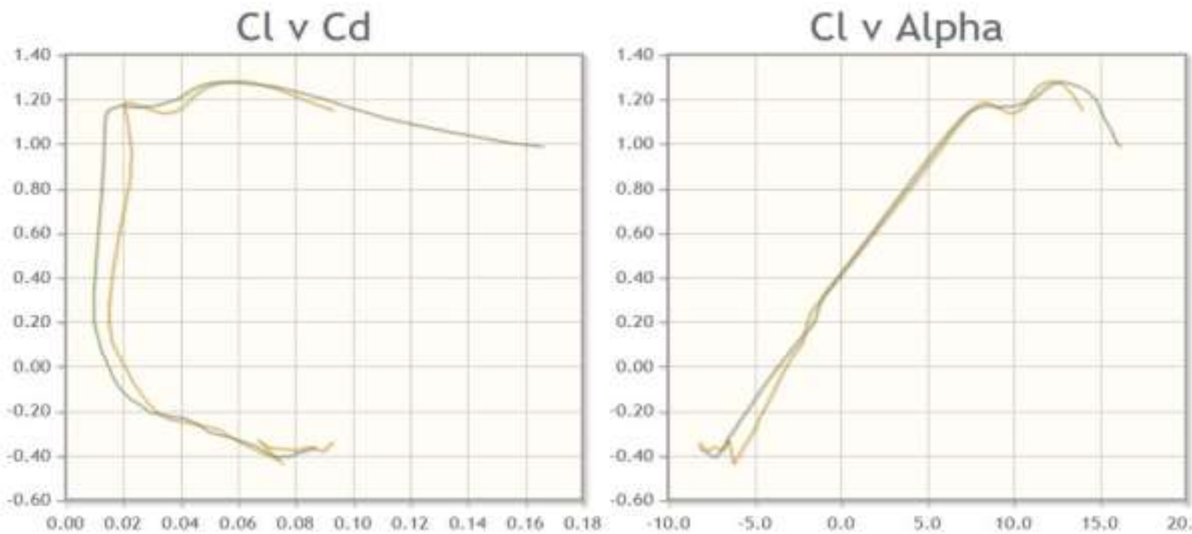


Figure 5: C_L vs C_D and C_L vs Alpha for E193 airfoil

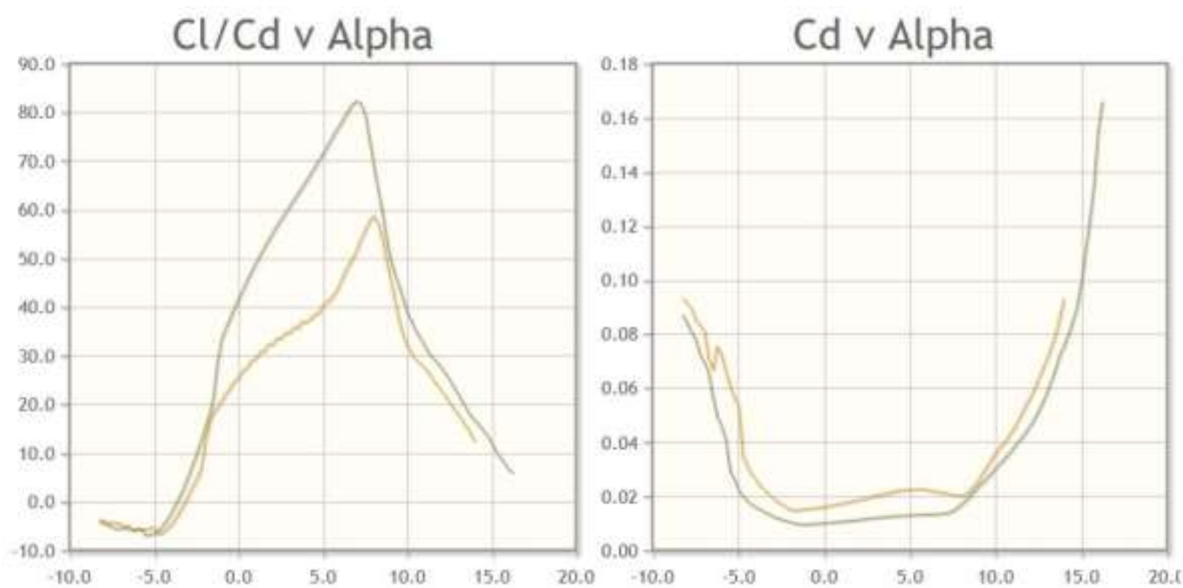


Figure 6: C_L/C_D vs Alpha and C_D vs Alpha for E193 airfoil

- **NACA 2410:**

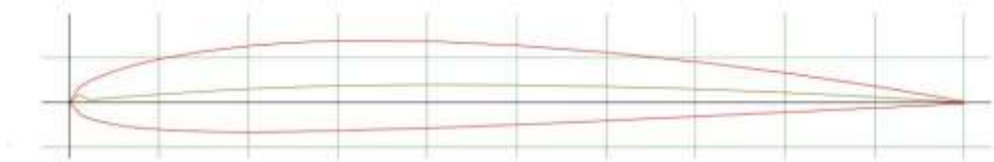


Figure 7: NACA 2410 airfoil shape

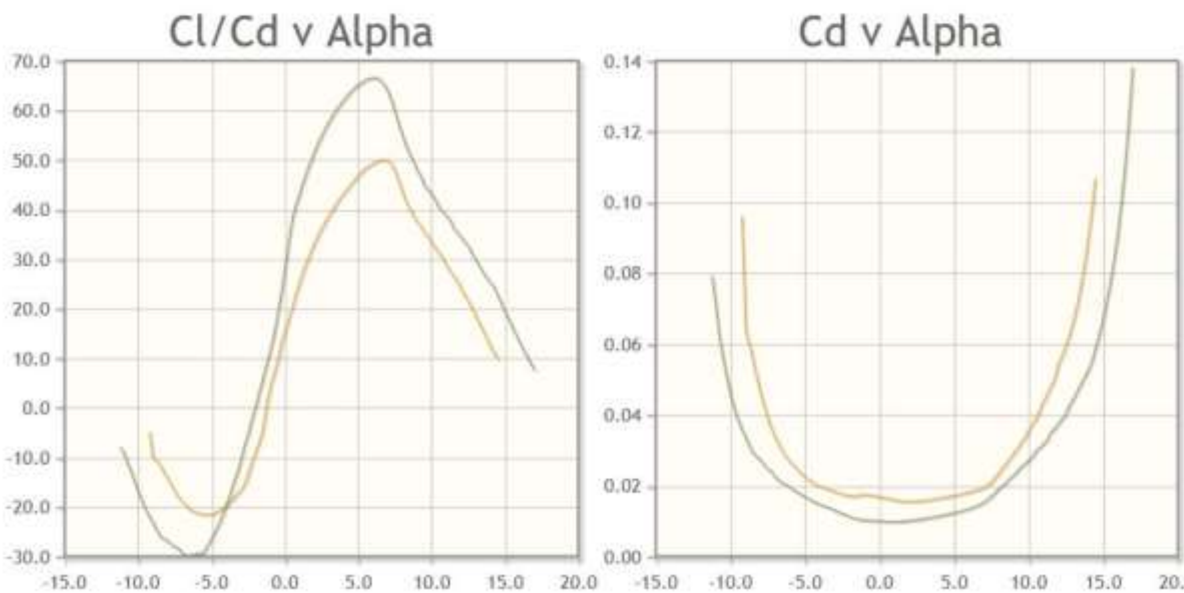


Figure 8: C_L vs C_D and C_L vs Alpha for NACA 2410 airfoil

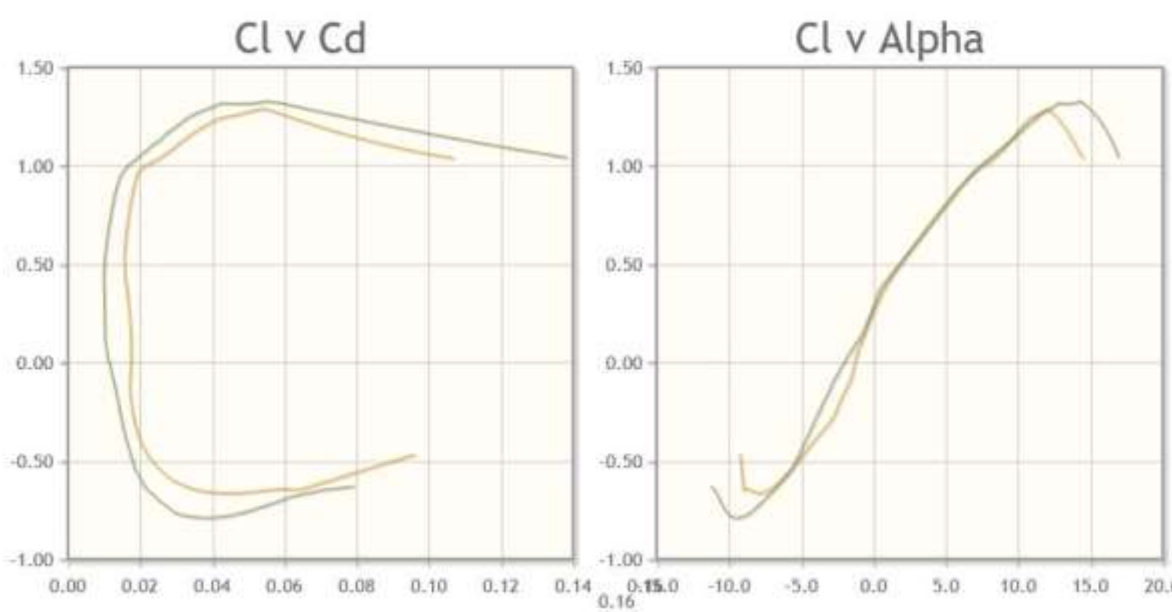


Figure 9: C_L/C_D vs Alpha and C_D vs Alpha for NACA 2410 airfoil

The results from the above comparison are shown in the table below:

Table 2: Characteristics of various air foils:

Airfoil	C_l at 0^o	C_{lmax}	C_d at 0^o	$(C_l/C_d)_{max}$	Stall angle
S1223	1.17	2.022	0.02337	54.5	12^O
E193	0.4114	1.2804	0.01614	58.7	12.25^O
NACA 2410	0.2623	1.2187	0.0158	50	13^O

Inference:

From the above observation of various characteristics, S1223 is chosen to be the suitable airfoil for our main wing. S1223 is a highly cambered airfoil. It produces high lift at lower angles of attack. Also, the manufacturing of S1223 is easy. The angle of incidence is chosen to 0^0

4.1.2. Aspect Ratio:

Aspect ratio of a plane wing is the ratio of its span and its mean chord. Wingspan is determined by deciding the aspect ratio of the wing. A wing with high aspect ratio will have less induced drag.

Based on literature survey and historical evidence, we chose the aspect ratio of our plane to be 7.5.

4.1.3. Angle of Incidence:

For a fixed wing aircraft, angle of incidence can be defined as the angle between the chord line and a reference axis along the fuselage. The angle of attack is the angle at which the wing meets the oncoming air. If the wing is tilted upward, it creates more lift. On the account of high lift produced by S1223, our wing has a zero angle of incidence.

4.1.4. Wing Planform and Configuration:

The shape of the wing, when viewed from top of the plane is called is called planform. Wing configuration for a fixed wing aircraft is its arrangement of lifting surfacerelative to other surfaces. Apart from the aerodynamic characteristics, ease of manufacturing and structural factors was also to be considered.

High wing configuration is most preferred for a slow flyer. Also high wing configuration prevents damages to wing while landing. Dihedral angle is added to the wing to give rolling stability to the aircraft. By referring to research article, we inferred that a rectangular wing can cause more rolling moment than a tapered wing. Getting inspired from the Eagle Flight we used biomimicking and Gull Wing configuration is adopted for our plane. Gull wing is a wing configuration with a prominent bend in the wing inner section toward the wing root. Advantages of Gull Wing:

i.) Good Glide characteristics

ii.) Gull wing induces rolling instability and helps in attaining high performance characteristics.

iii.) Sharp turns can be achieved i.e., turn rate is more.

From the above study, the wing configuration is chosen to be rectangular, gull wing with a dihedral of 10^o. High wing configuration was chosen.

4.1.5. Wing dimension Calculations:

During cruise condition,

Weight of the Aircraft = Lift of the Wing

The weight of the aircraft along with the payload was chosen to be 2.7 kg (1.2 + 1.5 kg)

Thus, Lift of the Aircraft = 2.7 *9.81 = 26.487 N

We also know that.

$$\text{Lift of the Aircraft} = ½*\rho*V^2*S*C_l$$

Thus, $$½*\rho*V^2*S*C_l = 26.487 \text{ N}$$

Here we know $\rho = 1.225$ kgm^{-3}, V= 12 ms^{-1}, C_l =1.035

Substituting these values in the lift equation, we get,

½*1.225*10^2*S*1.8 = 26.487 N

Thus, S = 0.30 m^2

We have chosen aspect ratio to be 7.5

Thus, b^2/S = 7.5

So, wingspan, b = 1.549 m

And, chord length c = 0.206 m

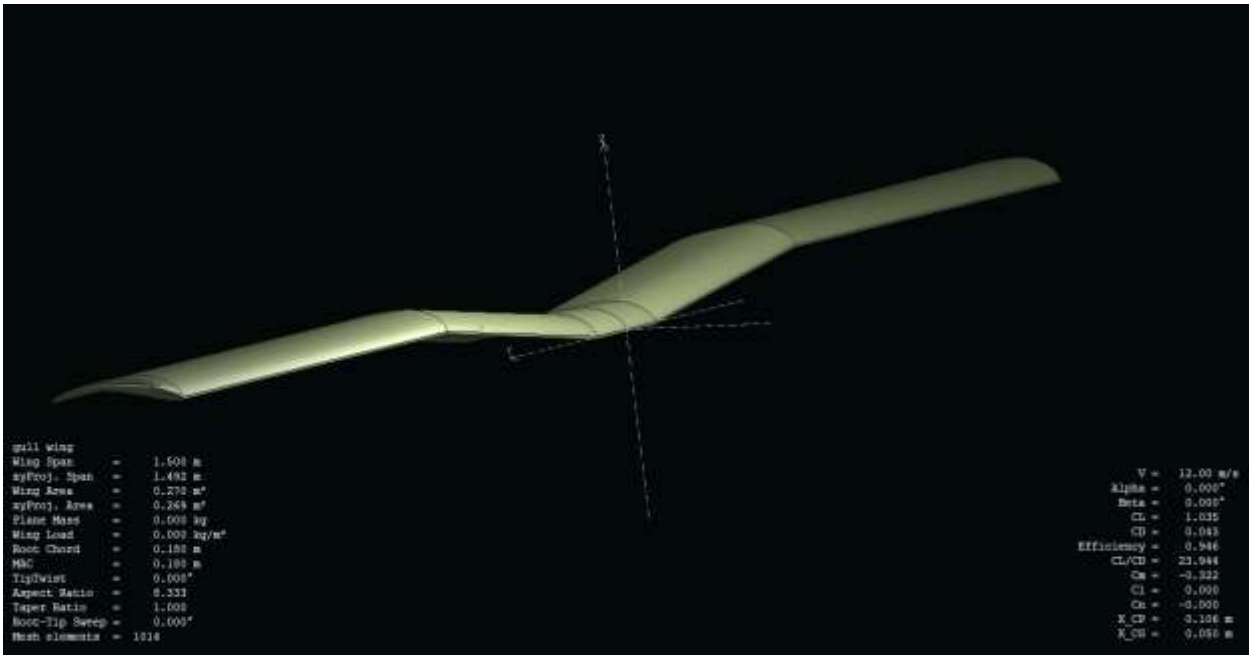


Figure 11: Wing analysis in XFLR5 Software

4.2. Body design:

The body of an aircraft is also called as fuselage. It greatly varies in size and shape depending upon the mission requirement. The different structures of fuselage include:

- Monocoque
- Truss
- Semi monocoque

A monocoque fuselage has its skin holding the fuselage structure together while the semi monocoque structure has both. For our model we have used monocoque structure. The bulk heads provide the necessary strength and stiffness to the skin. The payload bay is to be kept under the wing at the CG of the plane. Fuselage is designed in SOLIDWORKS and tested/simulated in ANSYS Fluent to ensure that there is a smooth flow over it. The length of the fuselage is calculated using the data (Relation between the span and the length) that we have got from literature survey.

4.3. Empennage design:

The tail (Empennage) part is attached with to the belly with the help of rod. The rod acts as reinforcement for the belly and so extra stiffeners for the fuselage can be avoided. Two slots are made at the end of the rod to hold the horizontal and vertical stabilizers. The stabilizers are sized in proportion with wing area based on the standard ratios we got from literature survey. The calculations are as follows:

i.) Horizontal Stabilizer:

Area of the Horizontal Stabilizer (S_h) = 25% of Wing Area = 0.25*0.3

S_h = 0.075 m^2

Width of the Horizontal Stabilizer = 0.35 m

Length of the Horizontal Stabilizer = 0.255 m

ii.) Vertical Stabilizer:

Area of the Vertical Stabilizer (S_v) = 10% Wing Area = 0.10*0.3

S_v = 0.03 m^2

Length of the vertical Stabilizer = 0.18 m

Height of the vertical Stabilizer = 0.19 m

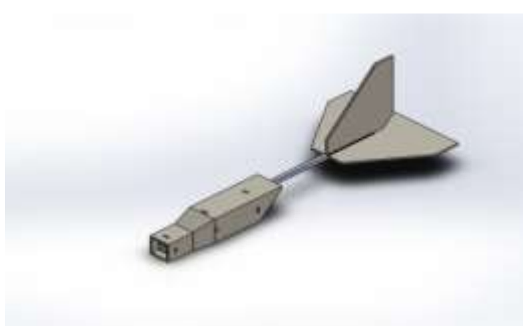


Figure 12: Fuselage and Empennage designed in SOLIDWORKS

4.4. Payload Bay Design:

The given dimensions of the payload bay are 5” x 1.5” x 1.5”. We have chosen to lift a weight of 1.5 kg. It was a great challenge to find a material that could weigh 1.5 kg within the given payload bay dimensions. After going through densities of various materials we found tungsten to have sufficient density to fulfill our payload weight requirement. But due to its high cost, copper was chosen which had lower density than tungsten but still satisfied our weight requirement.

The calculations for the mass of copper that can be sealed in the payload bay are as follows:

Density of copper	=	8960 kgm^{-3}
Volume of Payload Bay	=	5*1.5*1.5 = 11.25 $(inch)^3$
	=	$1.844 * 10^{-4}$ m^3
Mass of payload that can be accommodated	=	1.65 kg

The payload bay is kept inside the fuselage in such a way that the centers of gravity of plane and payload coincides.

5. Material Selection

The choice of the material to build the plane was done, considering all the constraints provided. It is stated that the empty weight of the plane must not exceed 1.5 kg. So the team carried out an elaborate survey to decide upon the materials to be used in fabricating the plane.

5.1. Fuselage:

The material used here must be capable of holding all the parts of the plane including electronic components and payload. The idea of using metal sheets to cover the fuselage was dropped owing to its greater weight and higher cost of constructing. After an intensive study, the team has decided upon using corrugated plastic sheets to cover the fuselage. All the bulkheads were made using corrugated plastic sheets. Aluminum was used in adding extra stiffness in the foremost bulkhead which serves as a motor mount.

5.2. Wing:

Wing is the most significant part in any aircraft. It should withstand all the loads of the aircraft. The material must be selected in such a way that it has more strength-to-weight ratio. After a detailed study we found two materials to fulfill our requirement, which are as follows:

i.) Balsa

ii.) High Density Foam

As far as properties of these materials are concerned, balsa is harder, heavier and more brittle than foam and foam is more tensile than balsa. It was found that reinforcing High Density Foam with some materials can increase its ability to withstand heavy loads. Hence we concluded to use High Density Foam in making our wing with aluminum tube used as a spar. Hot wire is used to get desired airfoil shape for the wing.

5.3. Horizontal and Vertical Stabilizers:

The Horizontal and Vertical Stabilizers are manufactured using corrugated plastic and is reinforced with cycle spokes.

6. Software Analysis

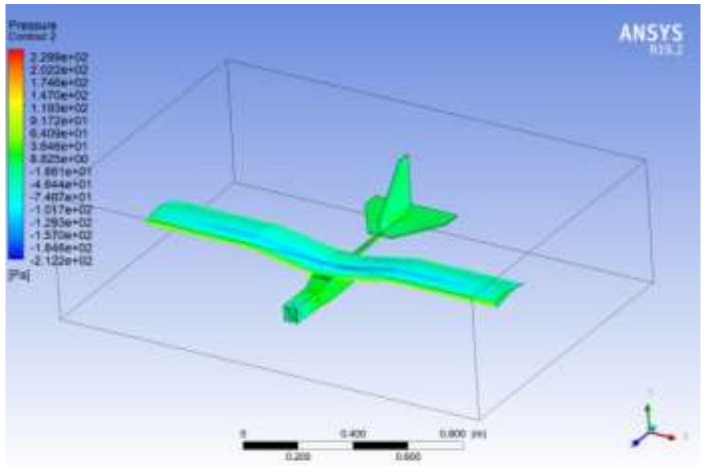
Following the modeling, analysis of our plane was done using variety of software (like XFLR5 and ANSYS). The analysis was done for various parameters whose discussion is as follows:

6.1. Pressure Distribution over the wing:

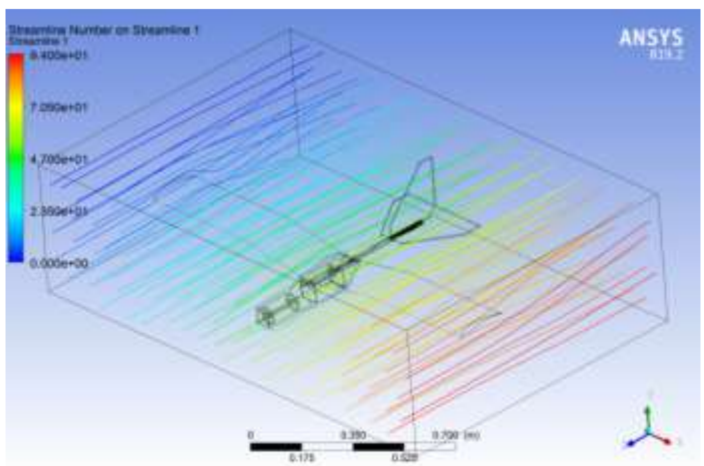
From the pressure distribution analysis, it is understood that wing root and wing leading edge and the nose of the fuselage are prone to more stress. Additional care was given in constructing these parts of the aircraft. Aluminum tube was used as reinforcement for the wing and Aluminium sheet was used as reinforcement for the motor mount in the fuselage nose. Materials were chosen considering the bending stresses causes due to pressure distribution.

6.2. Velocity Distribution over the plane:

The velocity profile over the plane shows that the flow is laminar and there are no turbulence and wake region. This is due to the slenderness of the plane. The fuselage is made as small as possible in order to get the optimum slenderness ratio.



(a)



(b)

Figure 14: (a) Pressure distribution over the wing (b) Velocity distribution over the plane

6.3. Analysis of Forces acting on the plane:

Force acting on the aircraft was analyzed using ANSYS and XFLR5, and the results were compared.

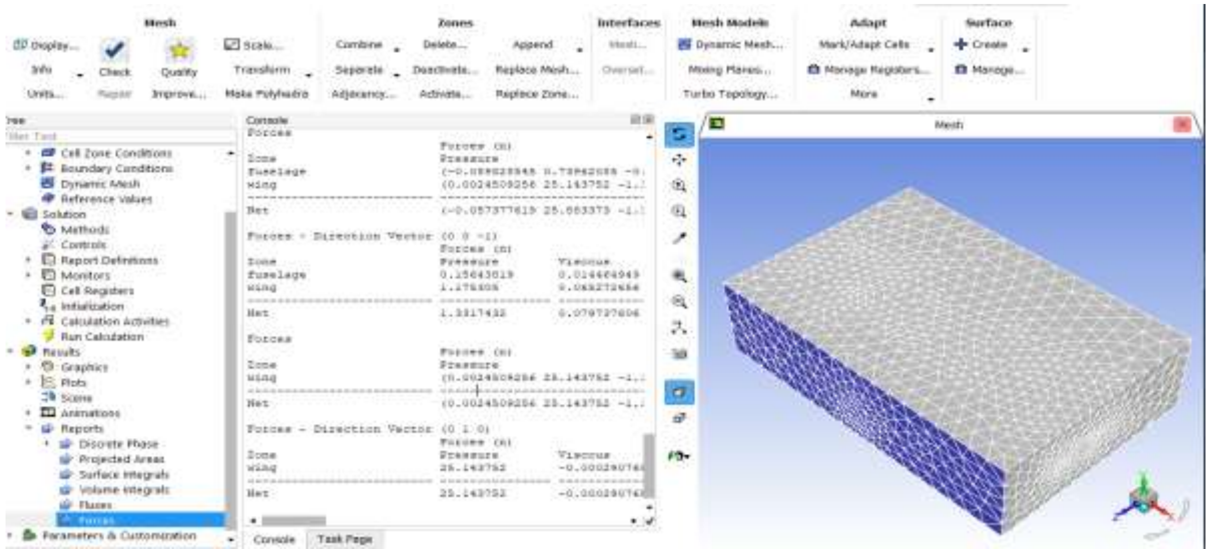


Figure 15: Analysis of Force in ANSYS

7. Power Plant Estimation

The chief requirement of the power plant is to have maximum possible thrust-to-weight ratio (T/W) which can be achieved either by increasing the thrust or decreasing the plane weight. After a thorough literature study, we have figured out thrust-to-weight ratios lying between 0.75 and 0.8 to be most desirable for our mission.

7.1. Battery Selection:

A minimum of two minutes flight is required to complete the task, considering the safety factor. To accomplish this power output, a three-cell battery was selected with 2200 mAh and 11.1 V (outburst voltage 12.6 V). Assuming the operational current as 25 A, a minimum power of 277.5 W will be produced.

7.2. Battery, ESC and Motor Considerations:

- Voltage of the battery cannot be greater than that the ESC can bear.
- Battery's output current has to be higher than ESC's rated current.
- Motor voltage is decided by ESC and ESC's is decided by battery's output.
- V_{MAX} of ESC should be less than that the motor can take in.
- ESC current should be greater than motor's current.

- Battery's discharge current should be greater than ESC's in order to get better performance.

7.3. Calculations:

Current drawn by motor	=	38 A	
Wattage of motor	=	570 W	
Current drawn by other electronics	=	5 A	
Total current drawn	=	43 A	
Battery discharge	=	35 C	
Battery charge	=	2200 mAh	= 2.2 Ah
So discharge due to electronics	=	43/2.2	= 19.545 C

which is less than the maximum discharge of the battery.

Flight time = 60 min / 19.545 = 3.07 min

= 2.46 min (considering safety factor- 0.8)

8. Thrust Calculation

Thrust is a force required to compensate for total drag and weight component of the aircraft. In an aircraft thrust is produced by a propulsion system, which in our plane is done by motor and propeller.

8.1. Motor and Propeller Selection:

The choice of motor and propeller plays a crucial role in attaining the estimated thrust-to-weight ratio. Exploring motors of different characteristics, we decided the motor C3536, 1050 Kv that weighs 116 g, to serve our demand. Propellers with various dimensions were also explored and one among the following three is to be selected. The RPM of the motor and speed of the aircraft were chosen to be 13230 and 12 ms^{-1} respectively.

Table 3: Propeller Dimensions and Thrust produced

Diameter (inch) x Pitch (inch)	Thrust produced (g)
11×5	1870

11 x 5.5	2100
12 x 5	2550

From the above observation it was inferred that propeller 11 x 5.5 suited our aircraft mission.

8.2. Thrust Analysis:

Propeller dimensions, Motor kV rating and its RPM are responsible for thrust production in an airplane.

The Kv rating of a motor is its measure of number of revolutions per minute that the motor turns when 1 volt is applied across it with no load attachment. A lower Kv motor produces more torque than a higher Kv motor because a low Kv motor has more winds of thinner wire—it will carry more volts at fewer amperes of current, producing more torque. Our motor has 1050 Kv rating and the RPM produced can be calculated as follows:

RPM of the motor = 1050 Kv * 12.6 V

(where, 12.6 is the outburst voltage of the motor)

= 13230

8.3. Dynamic Thrust:

Dynamic Thrust refers to the thrust produced in the aircraft when there is over it. A characteristic curve between thrust produced and aircraft speed to examine the thrust at various flight speed.

Dynamic Thrust was determined for various aircraft velocity (V_O) using the following formula

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

The thrust produced for various aircraft velocity ranging from (0-15) ms^{-1} was determined and graph was plotted using Excel Spreadsheet.

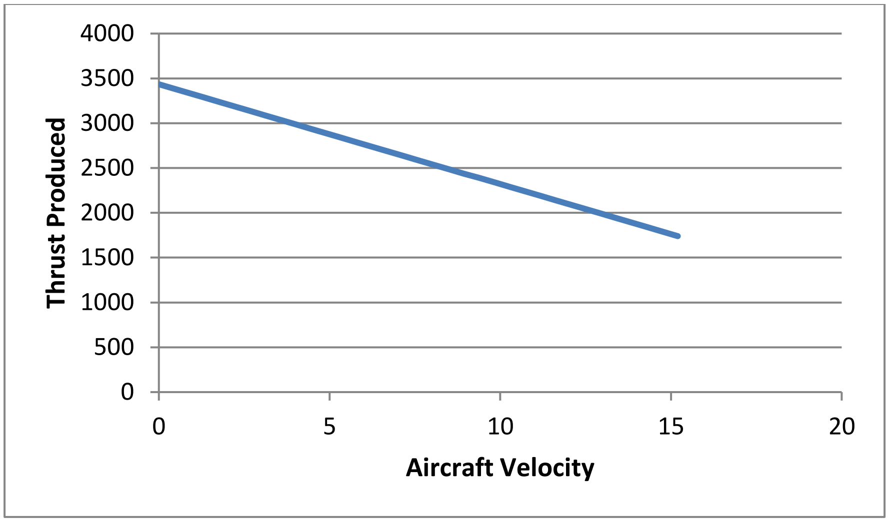


Figure 16: Thrust produced for different aircraft velocity

9. Drag Analysis

Drag is due to viscous nature of the fluid passing over the plane model. The main contributor for the drag is Skin Friction and a minimal amount from pressure (in case of no flow separation). If there is any flow separation on any surface of the plane then the wake region is formed and the pressure drag should be considered. Since the flow is at low velocities, the flow separation is minute. Thus pressure drag is least concern in our case.

The main contributors for total drag are Wing, Fuselage, Horizontal and vertical stabilizers. The following equation is used in order to find the drag characteristics of our plane.

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

where, C_{Do}= zero lift drag due to Fuselage + Wing + Horizontal and vertical stabilizers.

C_{lo} = Lift coefficient corresponding to zero lift drag for airfoil.

AR= Aspect Ratio

e= Span efficiency factor = 0.95

k= 1/ (π*AR*e)

Reference area, S_{ref} = 0.32 m^2

Table 4: Wetted surface area and planform area calculation

	Fuselage (m^2)	**Wing (m^2)**	**Horizontal Stabilizer (m^2)**	**Vertical Stabilizer (m^2)**
S_{wet}	0.11234	0.621	0.1378	0.07263
$S_{planform}$	0.0328	0.3	0.06627	0.001753
S_{wet}/S_{ref}	0.3511	1.941	0.431	0.2270

Wing:

Reynolds number R_e = $\rho vl/\mu$

= $0.2*1.225*12/ (1.81*10^{-5}) = 162430$ (laminar)

C_{Do} = $FF_w * C_f * S_{wet} / S_{ref}$

C_f = $0.664/ (R_e\text{^}1/2)$

FF_w = $1+ [0.6/(x/c)_m]*(t/c) +100(t/c)^4$

Fuselage:

Reynolds number R_e = $1.225*12*0.615/ (1.81*10^{-5}) = 499556$ (transition)

C_{Do} = $FF_F * C_f * S_{wet}/S_{ref}$

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

where, FR = Fuselage fineness ratio = Fuselage length / Fuselage diameter

= 0.765/0.08485 = 9.015

C_f = $1.328/ (R_e^{1/2})$

Horizontal Stabilizer:

Reynolds number R_e = $0.225*12*1.225/ (1.81*10^{-5}) = 182734$ (laminar)

C_{Do} = $FF_{hs} * C_f * S_{wet} / S_{ref}$

C_f = $0.664/ (R_e\text{^}1/2)$

FF_{hs} = $1+ [0.6/(x/c)_m]*(t/c) +100(t/c)^4$

Vertical Stabilizer:

Reynolds number R_e = $0.16*12*1.225/ (1.81*10^{-5}) = 129944$ (laminar)

C_{Do} = $FF_{vs} * C_f * S_{wet} / S_{ref}$

C_f = $0.664/ (R_e\text{^}1/2)$

FF_{vs} = $1+ [0.6/(x/c)_m]*(t/c) +100(t/c)^4$

Table 5: Zero-lift drag calculation

	Wing	Fuselage	HS	VS
C_{Do}	0.00438	0.0007285	0.0006822	0.0004276

$C_{Do\ (total)} = 0.006276$

$C_D = 0.006276 + 0.0446\ C_L^2 + 0.0446\ (C_L – 1.190)^2$

$C_{lo} = 1.190$ (from XFLR5)

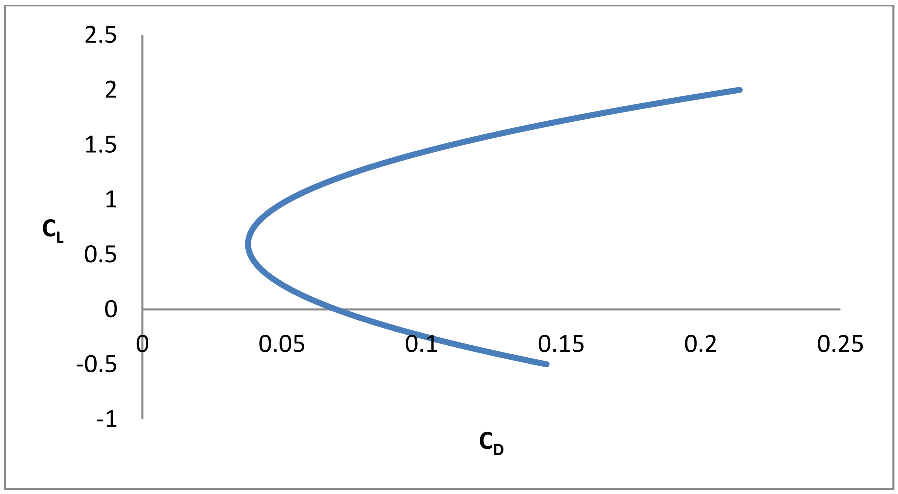


Figure 17: Coefficient of Lift for different Coefficient of Drag

10.Estimation of Centre of Gravity

Centre of Gravity is a point where the total weight of the plane is assumed to be concentrated. The location of center of gravity in a plane is crucial as it determines the aircraft's stability and performance. We used excel sheet to locate the center of gravity of our plane and compared the results with SOLIDWORKS.

Table 6: Overall Balancing Results

CG Location (x):	-10.52157
CG Location (y):	0
Neutral Point (x):	-5.150824
Static Margin:	-67.41%
All-Up Weight:	988.75
Total dCm/dα (rad^{-1}):	3.199508

Table 7: Wing Results

Overall

Configuration:	Monoplane
Planform Area:	347.0525
Wing Span:	47
Mean MAC:	7.967742
Aero Center:	-4.024194
Wing Center (z):	0
Mean AR:	6.064516
Mean λ:	0.55
Trailing Edge:	-26
0.25c Sweep (°):	8.172765

Table 8: Fuselage Results

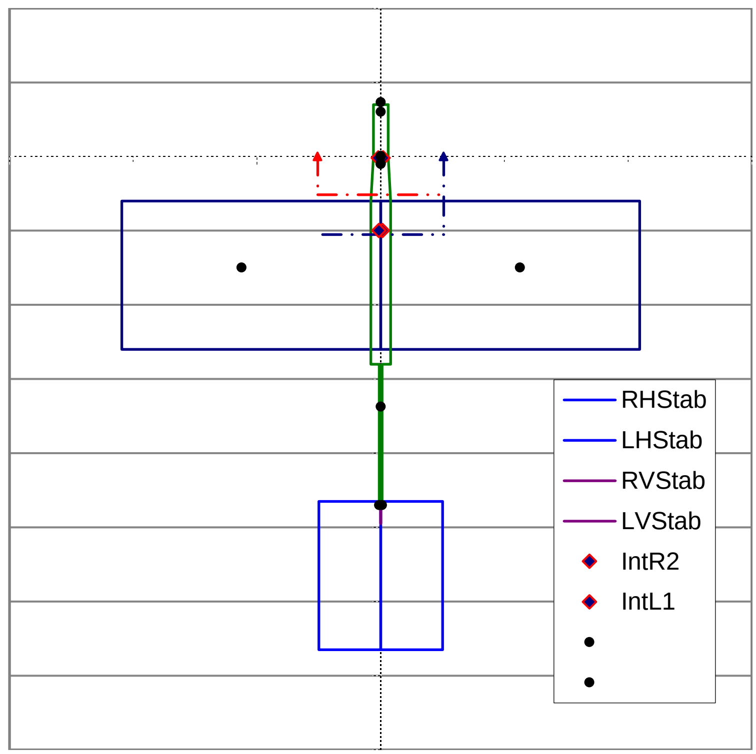
Length:	53.5
Width:	8
Area:	50.6875
dCm/dα (rad^{-1}):	0.020356

Table 9: Vertical Stabilizer Information

Configuration:	1
# of Stabilizers:	1
Total Area:	23.06977
l_t:	19.64479
V_v:	0.027784

Table 10: Horizontal Stabilizer Results

Configuration:	1
Planform Area:	58.63889
l_h:	25.40153
l_{h1}:	3.425726
l_t:	18.90416
Lift Slope (rad^{-1}):	3.771123
K_a:	0.120284
K_λ:	1.192857
K_h:	0.98995
dε/dα:	0.432626
V_h (tail volume ratio):	0.400878
η_t:	0.6
dCm/dα (rad^{-1}):	-0.51464



11.Test flight:

After the analysis, a scale down model of our aircraft was made. Using this model, flight tests were done. The end results were satisfactory.

11.Stability Analysis:

Static Stability of a plane along the longitudinal axis can be determined qualitatively using the graph C_{Mcg} **vs** α. It depends on two factors

- The slope of the graph should be negative i.e., ($dC_{Mcg}/d\alpha$ is -ve).
- Moment Coefficient at zero absolute angle of attack should be positive i.e., C_{Mo} should be +ve.

For the plane to be in balanced condition the necessary condition is the absolute angle of attack of the wing should lie in between Zero lift angle and the Stalling angle. Since the angle of attack given to our plane wing is zero it lies between Zero lift angle (around -4.5^o**)** stalling angle (12^o) the plane is in Balanced Condition. Static stability can be achieved by deciding the value of V_H and the value should be large enough. Once the V_H is fixed the remaining data required to solve the equation given below can be obtained from Excel sheet Cg calculations.

Equation for slope of the Graph:

$$dC_{Mcg} / d\alpha = a [h- h_{acwb} – V_H * (a_t / a) (1- d\varepsilon/d\alpha)]$$

$a = 2.49632$

$h = 4.5215$

$h_{acwb} = 4.024$

$V_H = 0.7$

$a_t = 3.771123$

$d\varepsilon/d\alpha = 0.4326$

On substituting all these values in the above equation we got

$$dC_{Mcg} / d\alpha = -0.1016$$

Since the value of $dC_{Mcg} / d\alpha$ is negative the required condition for static stability along longitudinal axis is satisfied and so the plane is statically stable.

12.APPENDIX

12.1. List of Symbols used:

C_L = Coefficient of lift for the wing.

C_D = coefficient of Drag for whole plane.

ρ = Density of air.

V = Free stream velocity

S = Span area of wing

AOA = Angle of Attack

AR = Aspect Ratio

C_L / C_D = Aerodynamic Efficiency

b = Span of wing

c = Chord of wing

$S_{planform}$ = Planform Area

S_{wet} = Wetted surface area

S_H = Horizontal Stabilizer Area

S_v = Vertical Stabilizer Area

C_{Do} = Zero Lift Drag Coefficient for the whole plane

C_f = Friction Coefficient

e = Span Efficiency Factor

a = Lift slope of whole plane

a_t = lift slope of tail

h = distance of Cg from the leading edge of the wing.

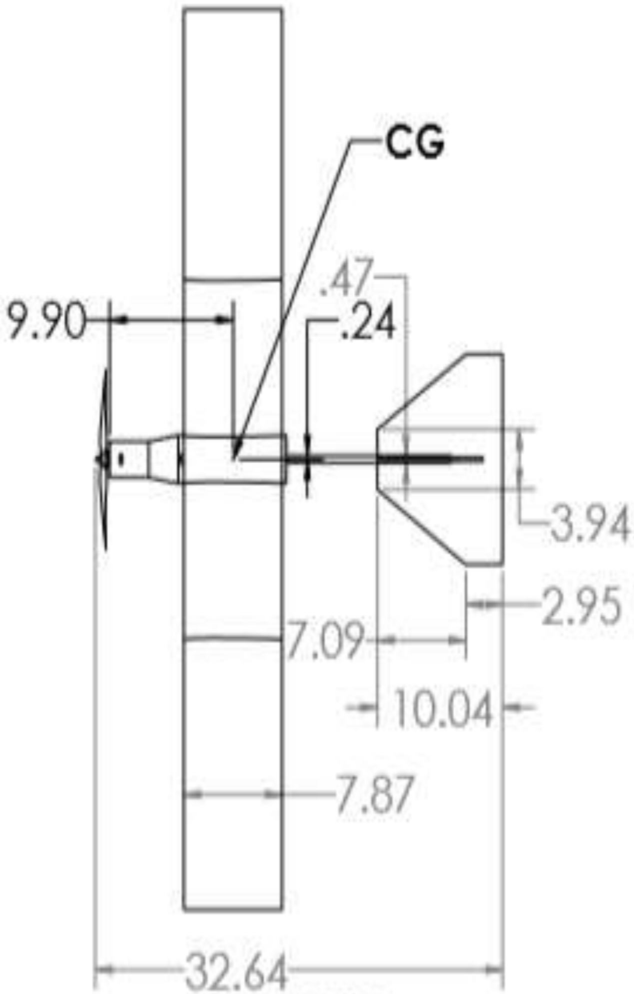
h_{acwb} = distance of aerodynamic centre from leading edge of the wing.

V_H = Tail volume ratio

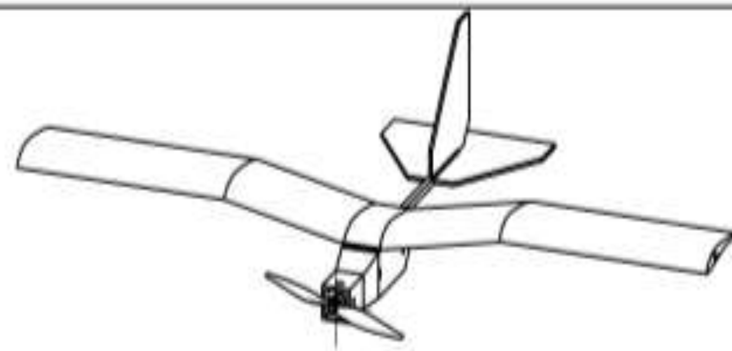
12.2. Aircraft Weight Build-up Schedule:

S No	Component	Weight (in gms)
1	Propeller	24
2	Motor and it's mount	120
3	Battery	190
4	ESC	45
5	Servos(x4)	36
6	Connectors and Extension Wires	30
7	Receiver	20
8	Fuselage	155
9	Wing	254
10	Payload Bay	40
11	Empennage	95
12	Adhesives(Glue and Tapes)	65
13	Extras Including Coverings	50
	Total	1124

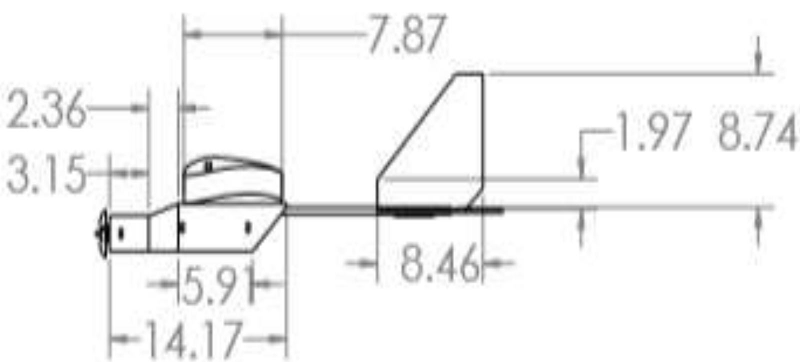
TEAM NAME	TEAM PEGASUS
TEAM ID	ADC2020113
COLLEGE NAME	SASTRA Deemed University
WEIGHT OF THE PLANE	1124 GRAMS
ALL DIMENSIONS ARE IN INCHES	



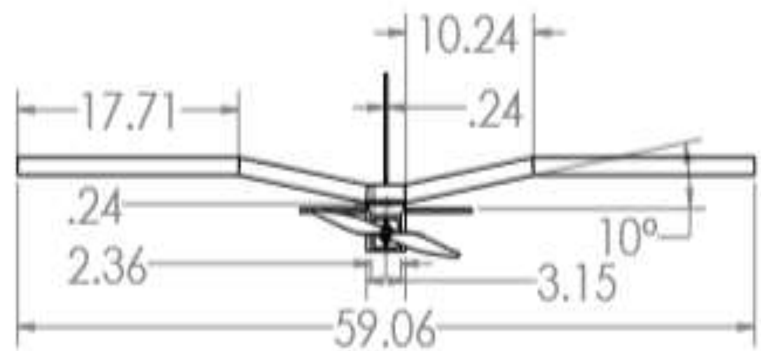
TOP VIEW



ISOMETRIC VIEW



LEFT SIDE VIEW



FRONT VIEW