Indian Institute of Technology Madras

Aerospace Engineering

Course:

Design of MAVs and UAVs

AS5213

Design of UAV for Forest Surveillance

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Chapter 1

MISSION SPECIFICATIONS

1.1 Mission Objective

From the data given by [6] the World Resources Institute, the forest fires results in 3 million more hectares of tree cover loss per year compared to 2001, an area roughly the size of Belgium and accounted for more than one-quarter of all tree cover loss over the past 20 years.
2021 was one of the worst years for forest fires since the turn of the century, causing an alarming 9.3 million hectares of tree cover loss globally, over one-third of all tree cover loss that occurred that year. Though down from the previous year, over 6.6 million hectares of tree cover was lost to forest fires in 2022, similar to other years over the past decade. And in 2023, the world has already seen heightened fire activity, including record-breaking burns across Canada and catastrophic fires in Hawaii.

The below bar graph gives Loss of trees in hectares due to fires compared to other reasons.

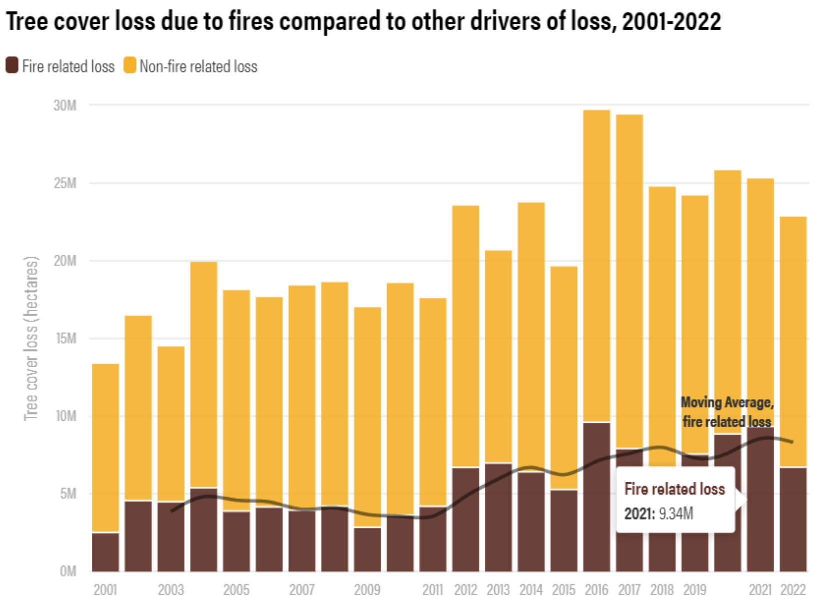

Figure 1.1: Tree cover loss[6]

The above mentioned is a fraction of problems we are facing regarding forest vegetation. There are lot of other problems as well such as smuggling, poaching, illegal deforestation etc.
Our goal is to design an UAV for Surveillance of Forest areas. Its primary task is to detect the start of forest fires which aids in resolving it as soon as possible. It can also be used to detect poachers, smugglers, help to monitor life forms in reserve forest, and can be used during rescue missions.

1.2 Mission Profile

The mission profile of the UAV is shown in the below figure.

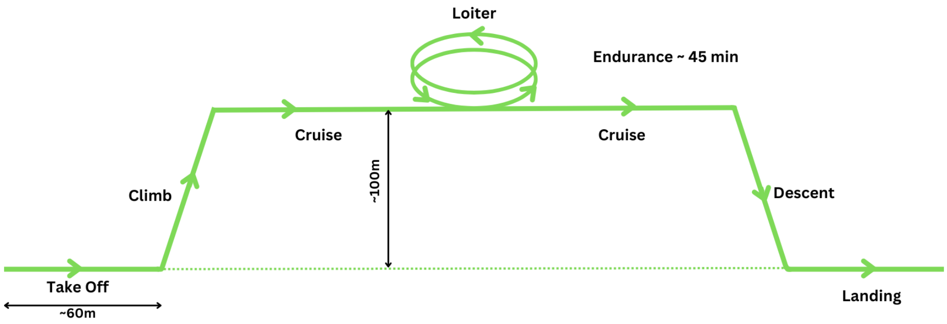

Figure 1.2: Mission Profile

- Takeoff: This stage has around 60m ground roll followed by a climb stage, The target altitude is 100m above takeoff point. The time for ground roll is around 10 s and for climb is around 1 minute.
- Air Time: The time of the flight in the air at the desired altitude is taken such that 90 percent of the time the UAV will be in a level turn, as in it will be loitering and scanning an area. The remaining air time is dedicated to cruise, which is the part of the mission where the UAV undergoes level flight and reaches the area to be scanned. The maximum air time is around 117 minutes.
- Landing : After the loitering, the UAV undergoes cruise and returns to the takeoff point where it undergoes a descent and then a ground roll and finally it stops. The descent will be for around 1 minute and the ground roll for about 10 seconds.

The takeoff altitude is assumed to be 7m above sea level (Chennai). The altitude of cruise is desired to be 107m above sea level.

By using the Li-Po battery for just a singe propeller, we estimate an endurance of 2 hrs. If we manage a cruise speed of 20 m/s, we can reach a theoretical range of around 144 Km.

1.3 Preliminary Configuration

The configuration adopted is a conventional fixed wing type UAV.

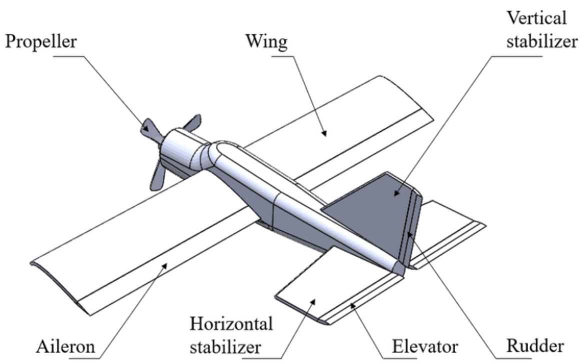

Figure 1.3: Fixed Wing configuration

The fixed wing configuration has various advantages:

- They generally have long flight times
- They can generally reach high airspeed
- They typically have a high coverage/range
- They can partially or completely glide for extended periods of time

Another option was to use a hybrid UAV. Reasons for choosing fixed wing over hybrid VTOL configuration are:

- Hybrid UAV configuration take up more energy as compared to fixed wing UAVs.
- Fixed wing UAVs have higher range and endurance which becomes useful for surveillance purposes.
- More number of rotors result in higher noise in some hybrid UAV configurations.
- The transition mechanism from VTOL to fixed wing mode in a hybrid UAV is extremely difficult to design as well as fabricate because of loss of lift during transition.
- The transition mechanism can be very costly.

1.4 Propulsion System

There are lot of Propulsion systems available commercially for a typical fixed wing UAV. A few commonly used propulsion systems are Electric propulsion (propeller and motor+ESC), Combustion engines (reciprocating engines), turbo engines, etc. There are other niche systems such as hydrogen cells, solar powered etc. Each propulsion system comes with their own advantages and disadvantages.[13] Based on the mission requirements and constraints, the best suited propulsion system is chosen for the UAV.
Considering the mission profile and the preliminary design of the UAV, It will not be weighing more than 12Kg approximately. This eliminates the need for any combustion type propulsion system, both reciprocating engines and turboprop/turbojet engines. Although they have high thrust to weight ratio, they do it at the cost of the weight of the UAV. Since a UAV which will be used for disaster management and surveillance for civilian use, it will not require extensive amount of power and speed, and hence there is no pressing need for such kind of systems. Hence the right propulsion system for the UAV would be an Electric Propulsion System.
Using an Electric Propulsion System for a UAV has a lot of advantages from manufacturing to operation. Since it has fewer moving parts compared to combustion engines, they are easy to manufacture and maintain. Using an electric Propulsion System also makes it easier for Autonomous integration in the future, as it is much easier to control. They also have very less vibration compared to a combustion engine propulsion system.[3] Apart from this, Electric Propulsion system would produce less emissions which is advantageous from an environment point of view. Also, noise levels is considerably less making it an advantage for surveillance UAVs[13]
A typical Electric propulsion system for a UAV consists of a brushless motor propeller combination, where the exact dimensions of the propeller is yet to be decided based on the mission requirements, a mount to fix on the wings or on the nose of the UAV based on the configuration, an ESC, or an Electronic Speed Controller which acts as the thrust control for the UAV, along with a Li-Po battery to power the components.[2] Generally any UAV's use brushless motors since they have relatively less friction between the rotating components, the losses are less compared to other motors like a typical DC motor. ESC's are one of the important components of the system. They control the rpm of the motor, so that we can change the amount of thrust produced by the engine(s).[14]

1.5 Previous Aircraft Study

These are some of the existing fixed wing UAVs with similar mission profiles.

(a) Believer UAV

(b) Raven B UAV

(c) Talon GT UAV

(d) PUMA LE UAV

(e) Bormatec Explorer UAV

(f) Albatross UAV

(g) Bormatec Maja UAV

(h) Sirius Pro UAV

(i) Mini Shark UAV

(j) Lockheed Martin Desert hawk

(k) AR3 Tekever

Figure 1.4: Previous Aircraft Study

Table 1.1: Table showing Data collected for previous Aircrafts considered

Sl No	Aircraft	MTOW (kg)	Empty Wt (kg)	Cruise Speed (m/s)	Wing Span (mm)	Wing Area (m^2)	Aspect Ratio	Battery
1	Believer UAV	5.5	2.4	20	1960	0.31	12.39	2× 14000mAh
2	Raven B UAV	2.2	1.82	9	1400	0.286	6.85	
3	Talon GT UAV	2.0	1.722	15	1000	0.14	7.14	Li-ion 4 S2P 14.4 V
4	Puma LE UAV	10.7			4600	0.68	13.6	
5	Bormatec Explorer UAV	4.0			2200			
6	Albatross UAV	10	4.4	19	3000	0.68	13.6	LiPO AS 8Ah
7	Bormatec MAJA	3.0			1800/2200			
8	Sirius PRO	2.7	1.7	18.06	1630	0.324	8.2	LiPo 800 gWt
9	Mini Shark UAV	5.5	2.8	15.28	2600	0.43	15.72	2100 mAh
10	Lockheed Martin Desert Hawk	4.0	2.0	25.56	1200	0.22	6.45	1.5 kg Battery
11	AR3 Tekever	22	14	33.33	3200			

Refer Appendix B for more details of each aircraft.

Chapter 2

FIRST WEIGHT ESTIMATE

2.1 Introduction

After getting the mission requirements, the next crucial thing is to get a better weight estimation. The weights of the aircraft are divided as follows

- **Crew Weight** W_{crew}

  It is the weight of the people boarded the plane. For commercial aircraft, the crew are passengers, and for military aircraft the crew are the soldiers.

- **Payload Weight** $W_{payload}$

  It is the weight of the module the aircraft is carrying - like for commercial aircraft the the payload will include passengers, baggage and cargo. For military aircraft the payload would be the missiles, bombs, surveillance units.

- **Fuel Weight** W_{fuel}

  It is the weight of the fuel carried by the aircraft for the whole mission. It will be decreasing with time in mission profile. For battery driven aircraft, the fuel is replaced with battery. There we considered W_{fuel} as $W_{battery}$.$W_{battery}$ will not vary over the mission profile, it remains the same.

- **Empty weight** W_{empty}

  It is the weight of the aircraft with structure, engines, avionics and landing gear. Basically, the weight of aircraft without crew, payload, and fuel.

The sum of these weights will give the total weight of the aircraft W_0. It is the Design Takeoff Weight (DTOW). W_0 will be the weight of the airplane at the instant of the starting of the mission. It varies during the mission, if the aircraft is fuel-driven or if the mission is payload drop.[4]

2.2 Payload

The payload is a camera used for the mission. A good choice of camera is Eagle Eye-30IE-M50 . The features include:

- High Control Accuracy Gimbal mounted camera.
- Q10T features 10x optical zoom and auto tracking function.
- 3-axis gimbal camera for UAV
- 0.492 Kg weight.
- Mainly supply in law enforcement, fire-fighting, power tower and pipeline inspection, search and rescue etc.

Figure 2.1: Camera

We assume overall payload to be less than 0.8 Kg including miscellaneous components and safety margins.

2.3 Battery

LiPo battery will be used as it is very common among UAVs which are electrically propelled. They have certain advantages over other type of batteries including high energy density, thin thickness, low internal resistance, high voltage, good safety etc. Further estimation of the exact characteristics of the battery used will be dealt with in a later section.

2.3.1 Preliminary C_D estimation

The coefficient of drag,

$$C_D = C_{D_0} + kC_L^2 \tag{2.1}$$

By taking data from [12], [9] and [9], the value of $C_{D_0]}$ is estimated to be 0.04 and the value of Ostwald's efficiency factor (e) is estimated to be 0.85. The average value of $\frac{L}{D}$ comes out to be 12.375. Similarly, the value of $C_{L_{max}} = 1.62$. We know the following relations:

$$k = \frac{1}{\pi e AR} \tag{2.2}$$

By substituting values, we get the value of k to be 0.047.
Hence, the drag polar is given by

$$C_D = 0.04 + 0.047C_L^2 \tag{2.3}$$

2.4 Method of Weight Estimation

The DTOW (W_0) is the sum of all the weights of the aircraft and beginning of the mission profile.

$$W_0 = W_{crew} + W_{payload} + W_{fuel} + W_{empty} \tag{2.4}$$

$$\implies W_0 = W_{crew} + W_{payload} + \frac{W_{fuel}}{W_0}W_0 + \frac{W_{empty}}{W_0}W_0 \tag{2.5}$$

$$\implies \boxed{W_0 = \frac{W_{crew} + W_{payload}}{1 - \frac{W_{fuel}}{W_0} - \frac{W_{empty}}{W_0}}} \tag{2.6}$$

Since we are dealing with as UAV which is battery driven, we can consider $W_{crew} + W_{payload}$ as $W_{payload}$, and W_{fuel} as $W_{battery}$. So the equation we end up with is

$$W_0 = \frac{W_{payload}}{1 - \frac{W_{battery}}{W_0} - \frac{W_{empty}}{W_0}} \tag{2.7}$$

The known term in this equation is $W_{payload}$. We need to find $W_{battery}$, the empty weight , W_e and W_0.
We need the relation between W_e and W_0 as shown below.

$$\frac{W_e}{W_0} = AW_0^L \tag{2.8}$$

For determining the constants A and L, we collect data of previous Aircrafts and make a log-log plot of $\frac{W_e}{W_0}$ vs W_0. Then, using linear regression, we fit a best fit curve to this data, hence determining the values of A and L.

Now, we relate the battery weight W_0 to the total weight W_0 by estimating total energy required. We use an iterative method to estimate the values of empty weight fraction and total weight using the above equations.

2.5 Total Energy Required

2.5.1 Ground Roll During Takeoff

The energy required for ground roll is given by the formula ([4] (eqn. 6.95)):

$$Energy_{groundroll_{takeoff}} T s_g = \frac{1.21W^2}{g\rho_{SL} SCL_{max}} \tag{2.9}$$

where, s_g is the distance along the ground moved by the UAV.

2.5.2 Climb

The velocity of climb can be found using the formula (ref):

$$V_{climb} = 1.2\sqrt{\frac{2W}{S\rho_{SL} C_{L_{max}}}} \tag{2.10}$$

We can find the average value of drag and hence power required for the climb is DV_{climb}.

$$Energy_{climb} = Power_{climb} t_{climb} \tag{2.11}$$

2.5.3 Cruise

We know that during Cruise,

$$L = W = \frac{1}{2}\rho V^2 SC_L \tag{2.12}$$

$$T = D = \frac{1}{2}\rho V^2 SC_D \tag{2.13}$$

$$Energy_{cruise} = Power_{cruise} t_{cruise} = D_{cruise} V_{cruise} t_{cruise} \tag{2.14}$$

2.5.4 Loiter

Loitering for a fixed wing UAV is essentially a level turn. We assume a bank angle of 30° for the purpose of calculation of Power required for loitering. The formula of thrust used is [4](eqn. 6.16):

$$T = D = \frac{1}{2}\rho V_{loiter}^2 S[C_{D,0} + K(\frac{2nW}{\rho V^2 S})^2] \tag{2.15}$$

Power required = TV = DV

$$Energy_{loiter} = Power_{loiter} t_{loiter} \tag{2.16}$$

Energy for descent is taken to be the same as that of climb and energy for ground roll during landing is taken to be the same as that of takeoff.

The total Energy required can be calculated by :

$$E_{Total} = E_{groundroll_{(takeoff)}} + E_{climb} + E_{cruise} + E_{loiter} + E_{Descent} + E_{groundroll_{(landing)}} \tag{2.17}$$

where,

- T and D are Thrust and Drag respectively.
- $n = \frac{L}{W}$

- ρ_X is the density of air at X.
- S is the wing area.
- C_L and C_D are Coefficients of Lift and Drag respectively.
- C_{D_0} is the parasitic drag.

2.6 Results of the Iterative Process

2.6.1 Code

Refer Appendix A for the script used.

2.6.2 A and L values

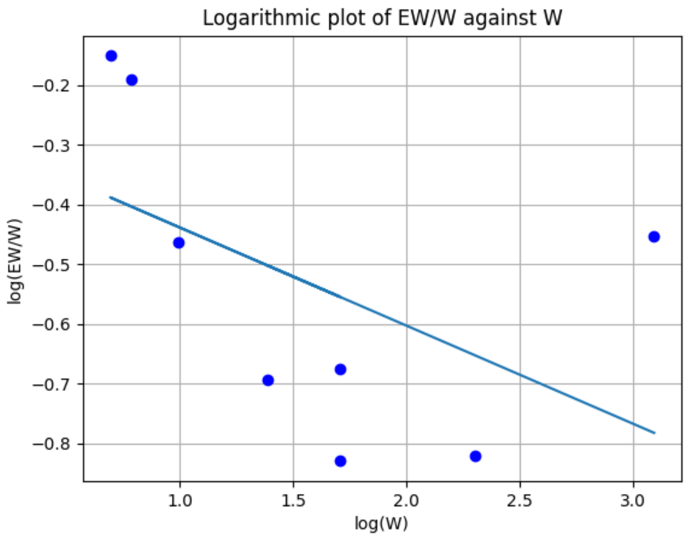

Figure 2.2: Linear regression to obtain values of A and L

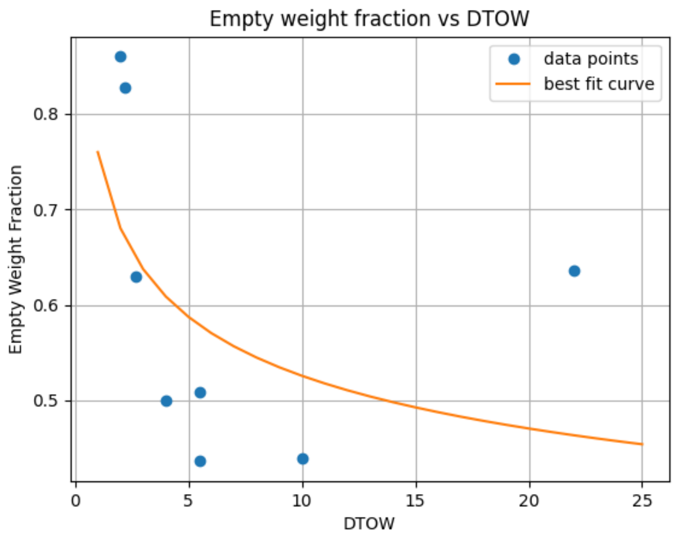

Figure 2.3: Curve fit using $\frac{W_e}{W_0} = AW_0^L$

The value of A = 0.76 and the value of L = -0.16 for the dataset we chose.

$$\frac{W_e}{W_0} = 0.76 W_0^{-0.16} \tag{2.18}$$

2.6.3 Battery Weight Estimation

Weight of battery, W_B is given by dividing the total energy for the mission divided by the specific energy (SE) of the battery used. We used an efficiency of 75% for the battery. We are considering the specific Energy of LiPo battery around 1.35 MJ/Kg.

$$W_B = 1.33\frac{E}{SE} = 1.33 \times \frac{1.14}{1.08} = \approx 1.4Kg \tag{2.19}$$

$$\frac{W_B}{W_0} = \frac{1.4}{5.95} = 0.235 \tag{2.20}$$

- The empty weight fraction is 0.572.
- The battery weight fraction is 0.235 and the weight of the battery is 1.4 Kg.
- The Design Take Off Weight is 5.95 Kg.

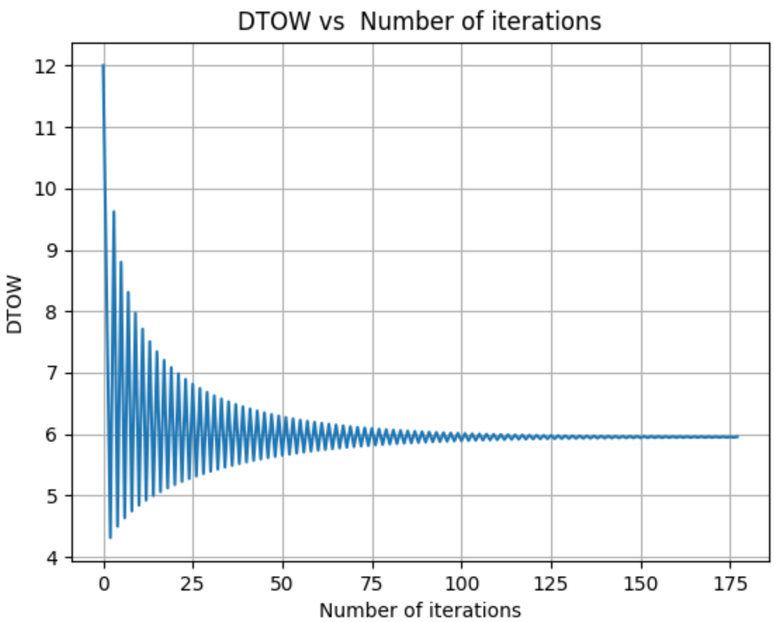

Figure 2.4: Plot showing the value of DTOW converging with iterations

The DTOW, W_0 converged at 5.95 Kg after 177 iterations.

After considerable research, an aspect ratio (AR) of 8 is chosen for the wing. The span is chosen to be 2.5 m. Rectangular planform is considered for the wing as it was found to have the best wing planform tradeoff ([1]).

$$AR = \frac{b^2}{S} \tag{2.21}$$

Hence, planform area,

$$S = \frac{2.5^2}{8} = 0.78m^2 \tag{2.22}$$

Chapter 3

$\frac{L}{Dmax}, \frac{T}{W}$ Estimation and Power plant Selection

3.1 Data Required for calculating $(\frac{L}{D})_{max}$

UAV	$\mathbf{A_{Top}}$ $(\mathbf{m^2})$	$\mathbf{A_{Side}}$ $(\mathbf{m^2})$	$\mathbf{S_{Wet}}$ $(\mathbf{m^2})$	$\mathbf{AR_{Wet}}$
Believer	0.144	0.168	0.5304	7.24
Talon GT	0.063	0.054	0.1989	5.03
Albatross	0.482	0.028	0.867	10.38
Sirius Pro	0.111	0.128	0.4063	6.54
Mini Shark UAV	0.11	0.103	0.3621	18.67
Desert Hawk III	0.045	0.103	0.2516	5.72

3.2 Calculation of Wetted Area , S_{Wet}

The method of calculation of S_{wet} is as given in [10] section 7.10.
We assume the wings and tail of the UAV is thin. So,

$$S_{wet_{wings_and_tail}} = 2.003 S_{exposed} \tag{3.1}$$

For estimating the wetted area of fuselage, the following relation is used:

$$S_{wet_{fuselage}} = 3.4(\frac{A_{top} + A_{side}}{2}) \tag{3.2}$$

where, A_{top} and A_{side} are the areas of the fuselage when seen from the top and the side. These areas were calculated by importing top view and side views of the reference aircrafts in AutoDesk Fusion360, calibrating the image appropriately, tracing the required areas and measuring the traced sketch.

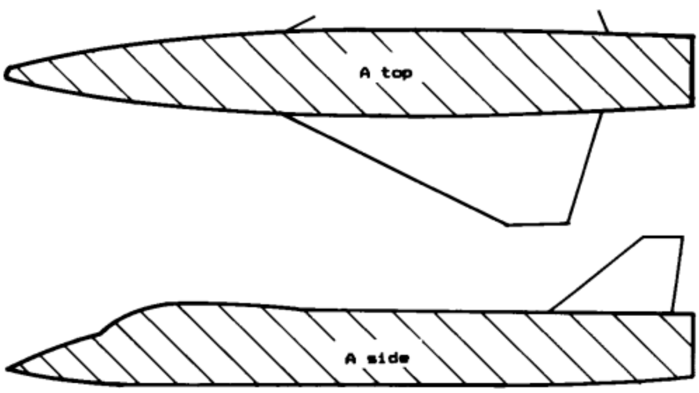

Figure 3.1: The top and side areas of fuselage as shown in [10]

3.3 Calculating The Zero-lift Drag $C_{D,0}$

This method uses the aircraft geometry which is known from the preceding design steps to estimate the zero-lift drag coefficient with the aid of an equivalent skin friction coefficient C_{fe}. The skin-friction coefficient multiplied with dynamic pressure and the wetted area gives the zero-lift drag:

$$C_{D,0} = q\ C_{fe}\ S_{wet} = q\ C_0\ S_{ref}$$

In contrast to the skin friction drag coefficient C_f, the equivalent skin-friction coefficient also includes other types of drag contributing to the zero-lift drag; such as form drag, interference drag, trim drag, and additional drag.

$$C_{D,0} = C_{fe} \cdot \frac{S_{wet}}{S_{Ref}}$$

3.3.1 Flat-Plate Skin-friction Coefficient

The factor depends on the amount of flow over the surface. For turbulent flow it is given by [10]

$$C_{f,e} = \frac{0.455}{(\log_{10} R_e)^{2.58} + (1 + 0.144M^2)^{2.58}}$$

Where R_e is the reynolds number evaluated as $R_e = \frac{\rho V_{cruise} l}{\mu}$.

3.4 Calculation of Ostwald Efficiency Factor

The Ostwald Efficiency factor, e, is approximated using the Aspect Ratio of the wing ([7])

$$e = \frac{1}{1.05 + 0.007\pi AR} \tag{3.3}$$

With an AR = 8, e = 0.816.

3.5 $(\frac{L}{D})_{max}$

The Lift-to-Drag ratio is the crucial design parameter. We need a good amount of lift, but with that it incorporates good amount of drag. We expect to get higher lift will lesser drag, which is maximum Lift-to-Drag ratio (L/D_{max}). So we need to find the L/D_{max} for our aircraft design. [12] is referred for the following derivation of $\frac{L}{D}$.

$$\frac{L}{D} = \frac{C_L}{C_D} = \frac{C_L}{C_{D,0} + KC_L^2}$$

For maximum L/D ratio, differentiate C_L/C_D with respect to C_L and set the value to zero.

$$\begin{aligned}
\frac{d\left(C_L/C_D\right)}{dC_L} &= \frac{C_{D,0} + KC_L^2 - C_L\left(2KC_L\right.}{\left(C_{D,0} + KC_L^2\right)^2} = 0 \\
\implies C_{D,0} + KC_L^2 - 2KC_L^2 = 0 \\
\implies C_{D,0} = KC_L^2 \\
\implies C_L = \sqrt{\frac{C_{D,0}}{K}}
\end{aligned}$$

Now, to find L/D_{max}, we need to find it in terms of $C_{D,0}$.

$$\begin{aligned}
\left(\frac{L}{D}\right)_{\mathrm{max}} &= \left(\frac{C_L}{C_{D,0} + KC_L^2}\right)_{\mathrm{max}} \\
\implies \left(\frac{L}{D}\right)_{\mathrm{max}} &= \frac{\sqrt{\frac{C_{D,0}}{K}}}{C_{D,0} + C_{D,0}} \\
\implies \left(\frac{L}{D}\right)_{\mathrm{max}} &= \frac{\sqrt{\frac{C_{D,0}}{K}}}{2C_{D,0}} \\
\implies \\
\implies \left(\frac{L}{D}\right)_{\mathrm{max}} &= \sqrt{\frac{1}{4C_{D,0}K}}
\end{aligned}$$

The above derivation of $L/D_{\max}$ is referred from [4] We find $C_{D,0}$ from Equivalent skin friction method ([10] (section 12.5)). The constant K is given as follows:

$$K = \frac{1}{\pi e AR}$$

where,
e = Oswald efficiency factor, AR is the Aspect Ratio.

3.6 Script for Estimating $\frac{L}{D}_{max}$ and $\frac{T}{W}$

Refer Appendix A for the script.

3.7 Plot of $(\frac{L}{D})_{max}$ vs $\sqrt{AR_{wet}}$ of previous Aircraft

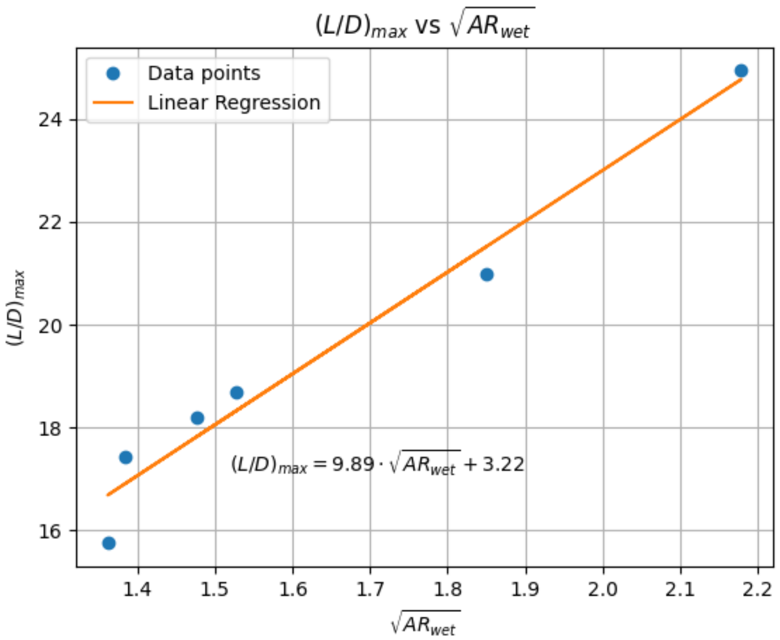

Figure 3.2: $(\frac{L}{D})_{max}$ vs $\sqrt{AR_{wet}}$

From the data collected, an average ratio of $\frac{S_{wet}}{S_{ref}}$ comes out to be around 3.674. Using this, we from the equation of the above plot, $(\frac{L}{D})_{max} = 9.89\sqrt{\frac{8}{3.674}} + 3.22 = 17.826$.
Since for a propeller aircraft the cruise is usually performed under $(\frac{L}{D})_{max}$, we can use the T = D and L = W criteria.
Hence, $(\frac{T}{W}) = \frac{1}{(\frac{L}{D})_{max}} = 0.056$.
Therefore, Required Thrust, T = 0.056 * DTOW = 0.056 * 5.95 * 9.81 = 3.27 N for Cruise.
Hence, the power for cruise , $P_{cr} = TV_{cr} = 3.27 * 20 = 65.4W$.

3.8 Energy and Power Calculation for Each Segment

3.8.1 During Ascent and Descent

During Take-off and Landing, Thrust to weight ratio is not exactly equal to $1/(L/D)_{max}$, since it is not a steady level flight condition like during cruise. Hence, we use a different formula to calculate the Thrust to Weight Ratio, which takes into account the drag coefficient at ground, lift coefficient for rotation, the friction coefficient of the ground and the take off distance. The expression is given by

$$(\frac{T}{W})_{ROC} = \frac{ROC}{\sqrt{\frac{2*W/S}{\rho\sqrt{\frac{C_{D0}}{K}}}}} + \frac{1}{(\frac{L}{D})_{max}} \tag{3.4}$$

where,
$(\frac{T}{W})_{ROC}$ = Thrust to Weight ratio during Climb,
ROC = Rate of Climb or Vertical Speed of the UAV,
ρ = Freestream density,
$\frac{W}{S}$ = Wing Loading

During the preliminary design of the mission profile, we require the UAV to climb 100 meters rough after 1 minute, we get the rate of climb as roughly 1.67m/s. adding a fluctuation of 20% of the rate of climb, we get the rate of climb as 2m/s. In the formula, the Thrust to weight ratio varies as the aircraft climbs since the weight of the aircraft reduces as it burns the fuel and the density is also a function of the altitude.
Although T/W constant, while designing we can take the maximum value of T/W ratio as thrust required during climb. The maximum thrust to weight ratio occurs when the density and wing loading are at maximum. Since they are at their highest value almost immediately after take-off we can assume ρ as 1.225 kg/m^3. DTOW of the aircraft = 7.826 Kg and The estimated planform area is 0.78 m^2 From this we can Estimate the

$$\text{Wing Loading, } \frac{W}{S} = \frac{5.95 \times 9.81}{0.78} = 74.83 N/m^2$$

Substituting these values in equation (24),

$$(\frac{T}{W})_{Climb} = \frac{1}{\sqrt{\frac{2*74.83}{1.2\sqrt{\frac{0.016}{0.049}}}}} + \frac{1}{17.826} = 0.124$$

Hence the Thrust required during the climb phase in the preliminary design,
$T = 0.115 \times DTOW = 0.124 \times 5.95 \times 9.81 = 7.24N$
Accounting for marginal errors, The thrust required during ascent can be approximated to 10N.
Take off Velocity for our UAV is going to be $V_{TO} = 1.2 \times V_{Stall}$

$$V_{Stall} = \sqrt{\frac{2W}{\rho S C_{L_{max}}}} \tag{3.5}$$

Substituting the known values in the equation, We get $V_{Stall} = 11.69 m/s$
$V_{TO} = 1.2 \times 11.69 = 14.08 m/s$
From These values, Power required during Ascent,
$P_{TO} = T_{TO} \times V_{TO} = 14.08 \times 10 = 140.8W$ $E_{TO} = 140.8 \times 60 \times 2 = 16,986J \approx 17KJ$.

3.8.2 Ground Roll

We know that the aircraft starts from rest. We assume that the aircraft accelerates uniformly to a lift-off speed , $V_{LO} = 1.1 V_{Stall}$ ([4] section (6.7)).
We assume a twin propeller configuration. So from [8], the maximum lift coefficient, $C_{L_{max}}$ can be assumed to be 1.2. Hence, from equation 25, we get the V_{stall} to be 11.69 m/s. Hence, $V_{LO} = 12.86 m/s$.
From the preliminary mission profile, we know that the time for ground roll is around 10 s. Hence the acceleration for the ground roll, $a = \frac{12.86-0}{10} = 1.286 m/s^2$.
The thrust for ground roll , $T_{groundroll} = ma = 5.95 * 1.286 = 7.65N$. The power can be estimated by using an average.

$$P_{groundroll} = \frac{\int_0^{V_{LO}} TV dV}{\int_0^{V_{LO}} dV} = \frac{1}{2} T V_{LO} = \frac{1}{2} * 7.65 * 12.86 = 49.19W \tag{3.6}$$

Energy = $49.19 \times 10 = 0.492 kJ$.
Same power, energy and thrust can be assumed for the ground roll during landing.

3.8.3 Loiter

We know that from [4] (section 6.2)

$$T_{Loiter} = D = \frac{1}{2}\rho V_{loiter}^2 S[C_{D,0} + K(\frac{2nW}{\rho V^2 S})^2] \tag{3.7}$$

where, n = $\frac{L}{W} = \frac{1}{cos(\phi)}$. ϕ is the bank angle which we assumed to be 25°. Substituting values into the formula, we get $T_{Loiter} = 4.72N$.
Power required for loitering, $P_{Loiter} = T_{Loiter}V = 4.72 * 20 = 94.4W$.

3.9 Total Energy Required

Stage	**Time**	**Thrust (N)**	**Power (Watt)**	**Energy (KJ)**
Ground Roll	10 s (for either)	7.65 (For either)	49.19 (for either)	0.492
Ascent+ Descent	120 s	10 (for either)	140.8 (For either)	17
Cruise	11.7 mins	3.27	65.4	45.91
Loiter	105.3 mins	4.72	94.4	596.42

We see that the total energy required is :

$$E_{Total} = 0.492 + 17 + 45.91 + 596.42 = 659.822KJ. \tag{3.8}$$

Estimating Battery Weight, W_B with the improved energy estimate and an overall efficiency factor of 75% and energy density of lipo as 0.8 KJ/Kg,

$$W_B = 1.33\frac{E}{SE} = 1.33\frac{0.659}{0.8} = 1.095kg \tag{3.9}$$

Including safety margin and additional weight, we take battery weight, $W_B = 1.25Kg$.

3.10 Choosing Battery

Based on the updated calculation of energy, we choose the battery. We estimate the battery based on the total energy required which came out to be 661.305 KJ. The Voltage rating of the battery multiplied by the mAh value gives the energy available in one cell of a battery. Multiply this by the number of cells times the number of sets to get the total energy stored in the battery.

1) Orange NMC 18650 11.1V 10000mAh 3C 3S4P Li-Ion Battery Pack

Figure 3.3: 3S4P Li-ion battery

- Weight = 787 g.
- Total Energy = $3(cells) \times 4(sets) \times 10 \times 11.1 = 1332$ Wh = 4795.2 KJ which is more than capable of running the mission.
- Maximum power by the battery = $voltage \times mAh \times discharge\ rate = 11.1 \times 10 \times 3 = 333W$.

We estimate around 0.75 efficiency for the battery. So, slightly higher energy requirements is kept.

3.11 Powerplant Configuration

The powerplant selection is the next crucial step in designing of unmanned aerial vehicle. The type of powerplant we decided for having for this aircraft is Electric Propulsion. Since it is easily available out of shelf, compared to jet propulsion, which are expensive and requires complex engine design and control. The popular propeller power plant configurations are pusher and tractor configurations. But for this aircraft design we decided to have a tractor configuration. Following are the reasons for having a tractor configuration for this aircraft design:

- Tractor configuration has the propeller mounted at the wing leading edge, while pusher configuration has the propeller mounted at the wing trailing edge.
- The tractor configuration has better aerodynamic performance than the pusher configuration, as it benefits from the increased airspeed and decreased angle of attack due to the propeller downstream.
- The tractor configuration has higher stall angle, lift–curve slope, maximum lift coefficient, and aerodynamic efficiency than the pusher configuration. It also has lower zero-lift drag coefficient and more stable pitching moment.
- The propeller effect improves the wing aerodynamics in both configurations by increasing the lift and drag coefficients, delaying the stall angle, and promoting the flow reattachment. However, the propeller downstream has a stronger and more turbulent effect than the propeller upstream. [5]

After deciding the powerplant configuration we need to decide the number of powerplants to be required. We can use one rotor, but the problems that arise with that design are - oversized propeller, high RPM rotor, propeller tip could reach Mach 1, which could cause adverse drag. So we have shifted to twin rotor configuration on the wing. The demand of required power from each rotor could be easily achieved with twin rotors. The only thing we need to be careful while designing twin rotors is the position of powerplant on the wing and maintaining equal RPM throughout the flight. We can go even further and keep 3 rotors, but it will increase the weight and complexity in the design of aircraft. So we settled to have a twin rotor tractor configuration powerplant.

Figure 3.4: Twin-Propeller configuration

3.12 Motor and Propeller Selection

The motors are selected in such a way that they can easily provide a thrust and continuous power which is more than the maximum required by any stage of our mission at around 80% throttle.

1) U5 Power Type UAV Motor KV400 with P15*5 Prop-2PCS/PAIR

(a) U5 Power Type UAV Motor

(b) P15*5 Propellers

Figure 3.5: Possible combination of motor and propeller

Chapter 4

Wing Loading W/S

Wing Loading (W/S) is weight of aircraft divided by area of the reference (not exposed) wing. Wing loading affects Take off and Landing distances, stall speed, climb rate and turn performances. It determines design lift co-efficient and impacts drag through its effect upon wetted area and wing span. Wing Loading and thrust to weight ratio must be optimized to attain Maximum take off weight with Minimum wing area. As wing loading of the aircraft is the one of the preliminary design parameter, it needs to be considered at every segment of the mission profile.

4.0.1 Takeoff

The aircraft takes off at about 1.1 times the stall speed so that the takeoff lift coefficient is the maximum takeoff lift coefficient divided by 1.21 (1.1^2)([10](section 5.3.3)).

$$C_{L_{TO}} = \frac{C_{L_{max}}}{1.21} = \frac{1.2}{1.21} = 0.99 \tag{4.1}$$

mentions the wing loading for takeoff for a propeller driven aircraft is given by:

$$(\frac{W}{S})_{TO} = (TOP)\sigma C_{L_{TO}}(\frac{HP}{W}) \tag{4.2}$$

where, TOP is the takeoff parameter, σ is the ratio of density of air at takeoff altitude to the density of air at sea-level ($\sigma \approx 1$), $\frac{HP}{W}$ is the specific power required for takeoff.
TOP for the aircraft can be found from the following plot.

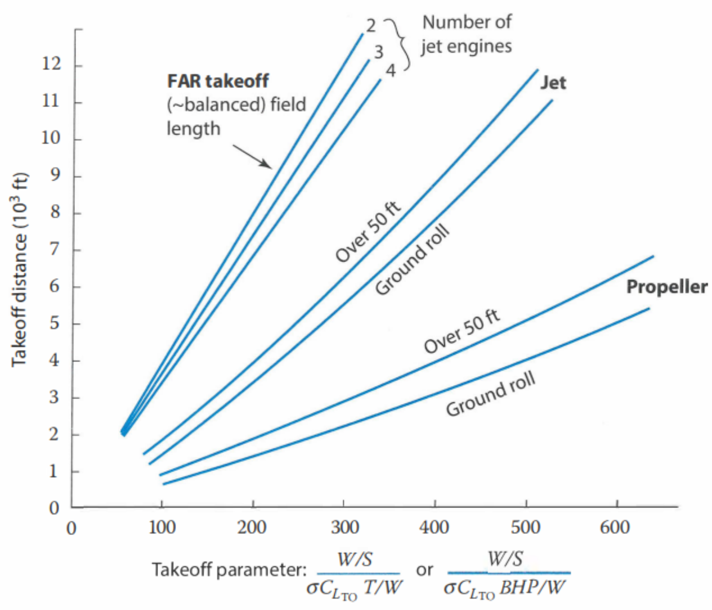

Figure 4.1: $W_{S_{TO}}$

We see that for a takeoff distance of the initially assumed 60m, we have TOP = 50 (approximately). Substituting the other values from previous sections,

$$(\frac{W}{S})_{TO} = 50 \times 0.99 \times \frac{0.029}{5.95 \times 9.81} = 0.025 N/m^2 \tag{4.3}$$

which is abnormal. So, using the formula:

$$(\frac{W}{S})_{TO} = \frac{1}{2}\rho V_{TO}^2 C_{L_{TO}} = \frac{1}{2} \times 1.2 \times (\frac{15.38}{1.1})^2 \times 0.99 = 170.01 \tag{4.4}$$

With a tolerance of 5%, we get:

$$161.51 N/m^2 < (\frac{W}{S})_{Loiter} < 178.51 N/m^2 \tag{4.5}$$

4.0.2 Climb

For climb, we can write $T = D + W sin\gamma$ and $L = W sin\gamma$.
$D = \frac{1}{2}\rho V_{cl}^2 S(C_{D_0} + KC_L^2)$ Using these, we get the formula for wing loading of climb:

$$\frac{W}{P} = \frac{\eta}{[\frac{\rho V_{cl}^3 C_{D_0}}{2(\frac{W}{S})} + \frac{2K cos^2\gamma}{\rho V_{cl}(\frac{W}{S})}]} \tag{4.6}$$

Where, $V_{C_L} = V_{TO} = 14.08$, climb angle, $\gamma = 10°$ and other quantities are known. Solving for $\frac{W}{S}$, we get $(\frac{W}{S})_{climb} = 240.85 N/m^2$. Adding a tolerance of 5%, we get:

$$228.81 N/m^2 < (\frac{W}{S})_{climb} < 252.89 N/m^2 \tag{4.7}$$

We assume the same wing loading for descent.

4.0.3 Cruise

A propeller-powered aircraft loses thrust as its speed goes up, so that it gets the maximum range when flying right at the speed for best $\frac{L}{D}$ ([10]). Therefore, to maximize range, a propeller aircraft should fly at a speed such that the parasitic drag equals the induced drag.

$$qSC_{D_0} = qS\frac{C_L^2}{\pi ARe} \tag{4.8}$$

During cruise, Lift equals the Weight, so that the lift coefficient equals the wing loading divided by the dynamic pressure.

$$C_L = \frac{W}{Sq} \tag{4.9}$$

Substitute into the previous equation, we get

$$\frac{W}{S} = q\sqrt{\pi AReC_{D_0}} \tag{4.10}$$

For cruise, $q = \frac{1}{2}\rho V_{cr}^2 = \frac{1}{2} \times 1.2 \times 400 = 240 Kg/ms^2$. Hence,

$$(\frac{W}{S})_{Cruise} = 400 \times \sqrt{\pi \times 8 \times 0.816 \times 0.02} = 153.71 N/m^2 \tag{4.11}$$

With a tolerance of 5%, we get:

$$146.02 N/m^2 < (\frac{W}{S})_{Cruise} < 161.39 N/m^2 \tag{4.12}$$

4.0.4 Loiter

The major part of our mission is loitering. Loitering for a fixed wing UAV is by means of sustained turn. For an aircraft which must be optimized for loiter, the wing loading should be selected to provide a high L/D. For a propeller aircraft, loiter is optimized when the induced drag is three times the parasitic drag according to [10] (Chapter 5), which results in:

$$\frac{W}{S}_{Loiter} = q\sqrt{3\pi(AR)eC_{D_0}} \tag{4.13}$$

For our UAV, we have an Aspect Ratio (AR) of 8, Ostwald Efficiency factor (e) of 0.816 and cruise velocity of 20 m/s.
$q = \frac{1}{2}\rho(20^2) = 0.5 \times 1.2 \times 400 = 240Kg/ms^2$. The value of C_{D_0} can be taken to be 0.02 based on previous aircraft study which agrees with [10] for a clean propeller aircraft.
Substituting the values, we get:

$$\frac{W}{S}_{Loiter} = 240 \times \sqrt{3 \times \pi \times 8 \times 0.816 \times 0.02} = 266.23N/m^2 \tag{4.14}$$

With a tolerance of 5%, we get:

$$252.92N/m^2 < (\frac{W}{S})_{Loiter} < 279.54N/m^2 \tag{4.15}$$

4.0.5 Stall

After the preliminary stalling condition, previously derived in Chapter 3, we need to estimate it again for getting the wing loading for Stall condition. As stated earlier, the value of $C_{L_{\max}}$ for a propeller-driven aircraft without any flaps is assumed to be 1.2. The cruise velocity, V_{Cruise}, is 20 m/s. [4] (section 6.8) mentions that the approach velocity of an aircraft is about $1.3 \times V_{\text{Stall}}$. Assuming the approach velocity to be nearly the same as the cruise velocity, we get: :

$$V_{Stall} = \frac{V_{Cruise}}{1.3} = 15.38 \tag{4.16}$$

[10](section 5.3.2) mentions that at stall:

$$W = L = q_{stall}SC_{L_{max}} \tag{4.17}$$

which gives,

$$(\frac{W}{S})_{Stall} = \frac{1}{2}\rho V_{Stall}^2 C_{L_{max}} = \frac{1}{2} \times 1.2 \times (15.38)^2 \times 1.2 = 170.31N/m^2 \tag{4.18}$$

With a tolerance of 5%, we get:

$$161.79N/m^2 < (\frac{W}{S})_{Stall} < 178.83N/m^2 \tag{4.19}$$

4.0.6 Maximum Speed

[1](section 6.5) mentions that the maximum allowable velocity, which is the high speed limit for an aircraft is close to $1.2 \times V_{Stall}$. Hence,

$$V_{max} = 1.2V_{Stall} = 1.2 \times 20 = 24m/s \tag{4.20}$$

The same conditions for cruise holds, but with a velocity of 24m/s.

$$\left(\frac{W}{S}\right) max \text{ speed} = \left(\frac{24}{20}\right)^2 \times \left(\frac{W}{S}\right) Cruise \tag{4.21}$$

With a tolerance of 5%, we get:

$$210.27N/m^2 < (\frac{W}{S})_{maxspeed} < 232.41N/m^2 \tag{4.22}$$

4.0.7 Landing

According to [10](section 5.3.2), the takeoff maximum lift coefficient that is about 80% that of the landing value. Hence,

$$C_{L_{Land}} = 1.25 \times C_{L_{TO}} = 1.25 \times 0.99 = 1.2375 \tag{4.23}$$

From [4](section 6.8), we get Touch Down velocity, V_{TD}.

$$V_{TD} = 1.15V_{Stall} = 17.69m/s \tag{4.24}$$

Hence,

$$(\frac{W}{S})_{Land} = \frac{1}{2}\rho V_{TD}^2 \times C_{L_{Land}} = 232.35N/m^2 \tag{4.25}$$

With a tolerance of 5%, we get:

$$220.73N/m^2 < (\frac{W}{S})_{Land} < 243.97N/m^2 \tag{4.26}$$

4.0.8 Maximum Ceiling

[10](section 5.3.11) gives the relation:

$$\frac{W}{S} = \frac{\left(\frac{T}{W} - G\right) \pm \sqrt{\left(\frac{T}{W} - G\right)^2 - \frac{4C_{D_0}}{\pi ARe}}}{\frac{2}{q\pi ARe}} \tag{4.27}$$

G is the climb radient. For maximum ceiling, the climb gradient, G can be set to zero to represent level flight at the desired altitude. The $(\frac{T}{W})$ for the climb is 2.2 W/N and other values are found from previous sections. Hence, for maximum ceiling,

$$\left(\frac{W}{S}\right)_{\text{max ceiling}} = \frac{2.2 \pm \sqrt{(2.2)^2 - \frac{4\times0.02}{\pi\times8\times0.816}}}{\frac{2}{240\times\pi\times8\times0.816}} \tag{4.28}$$

which gives abnormal values. So, we use a simple method to solve.
From [10](section 5.3.12), we get the wing loading for minimum power,

$$\left(\frac{W}{S}\right)_{\text{max ceiling}} = q\sqrt{\pi AReC_{D_0}} = 130N/m^2\,\mathrm{N/m^2} \tag{4.29}$$

which is similar to the value at cruise.
With a tolerance of 5%, we get:

$$123.5\,\mathrm{N/m^2} < \left(\frac{W}{S}\right)_{\text{max ceiling}} < 136.5\,\mathrm{N/m^2} \tag{4.30}$$

4.1 Wing Loading Range

Flight Condition	**Wing Loading** (N/m^2)
Takeoff	161.51 - 178.51
Climb and Descent	252.92 - 279.54
Cruise	146.02 - 161.39
Loiter	252.92 - 279.54
Stall	161.79 - 178.83
Maximum Speed	210.27 - 232.41
Maximum Ceiling	123.5 - 136.5
Landing	220.73 - 243.97

From these ranges, the lowest range of wing loading must be taken to ensure that the wing is large enough for all flight conditions ([10](section 5.4)). We are considering it to be that of cruise for a respectable size of wing.Hence, the range of Wing loading is 123.5N/m^2 - 136.5N/m^2. We choose 127.76 N/m^2 as our required Wing loading.

4.2 Wing Loading of Previous Aircraft Data

UAV	**Wing Loading** (N/m^2)
Believer	125.48
Talon GT	140.14
Albatross	144.26
Sirius Pro	81.75
Mini Shark UAV	125.48
Desert Hawk III	178.36

The average wing loading of the previous aircrafts come out to be 132.58 N/m^2, which is close to the value we have calculated.

Chapter 5

Second Weight Estimation

5.1 Method Used

From chapters 3 and 4, we know the thrust and wing loading for the UAV, for different phases of the mission segment. Using this knowledge, we can calculate the power and thrust required, from which we can find the battery, propeller and motor weights. With this information in hand, In our previous Weight calculation formula, we have one less unknown variable. In (2.6), we now know W_{fuel}.
The Second iteration of the weight estimate can now be done using the formula,

$$W_o = \frac{W_{crew} + W_{payload} + W_{battery}}{1 - \frac{W_{empty}}{W_o}} \tag{5.1}$$

The solution for W_o is found by iterating W_o just like the case of First Weight Estimate in Chapter 2, except this time, there is only one unknown value, the empty weight Fraction. The values of A and L are retained from the first weight estimate.

Refer Appendix A.1 for the script used.

5.2 Results of the Iteration

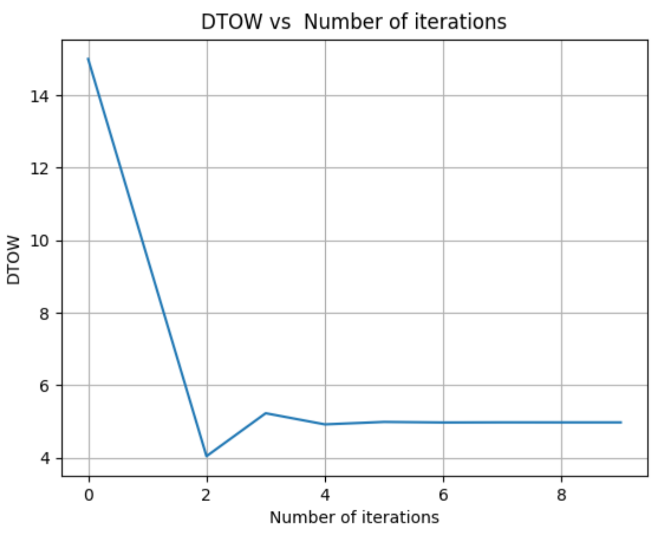

Figure 5.1: Plot showing W_0 converging

W_0 converged to 5.47 Kg after 9 iterations including a tolerance of 10%. Empty weight fraction is 0.62 and battery weight fraction is 0.23 and the payload fraction is 0.15.

Chapter 6

Wing Configuration

6.1 Airfoil - some key terms

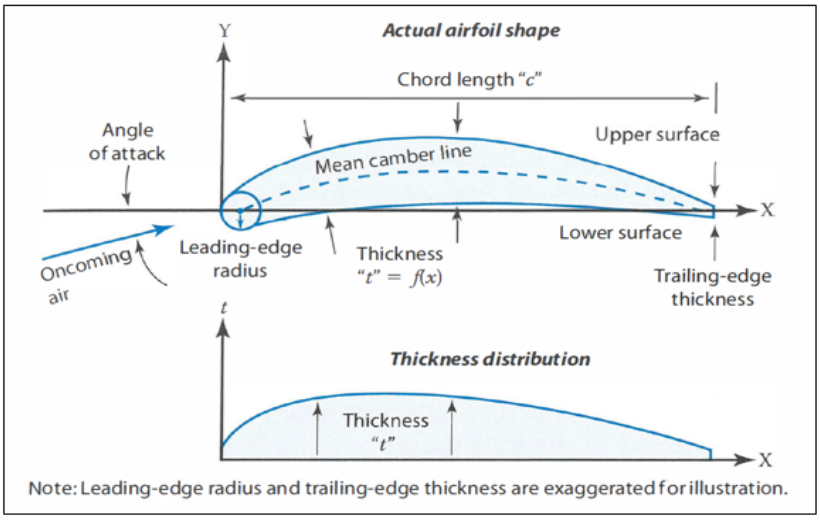

Figure 6.1: Taken from [10]

- **Airfoil Thickness Ratio** - The ratio between the maximum thickness or depth of an airfoil section and its chord length, expressed as a percentage of the chord length.
- **Wing Sweep** - The sweep angle of a wing is that between a line drawn along the span at a constant fraction of the chord from the leading edge, and a line perpendicular to the centerline.
- **Taper Ratio** - Wing taper ratio is the ratio between the tip chord and the centerline root chord.
- **Twist** - "Geometric twist" is the actual change in airfoil angle of incidence, usually measured with respect to the root airfoil."Aerodynamic twist" is the angle between the zero-lift angle of an airfoil and the zero-lift angle of the root airfoil.
- **Wing Incidence** - The wing incidence angle is the pitch angle of the wing with respect to the fuselage.
- **Wing Dihedral** - Wing dihedral is the angle of the wing with respect to the horizontal when seen from the front.

6.2 Airfoil Selection

6.2.1 Shortlisting Airfoils

First, get the data of $C_{L_{max}}$ and $\left(\frac{C_L}{C_D}\right)_{max}$ and go through different airfoils from the dataset of airfoiltools. Then, from the list of the airfoils, choose the airfoils that reaches the minimum requirement of the required

$C_{L_{max}}$ and $\left(\frac{C_L}{C_D}\right)_{max}$.
Based on our requirement we have chosen 5 airfoils:

- **NACA2412**
- **NACA2415**
- **NACA4412**
- **NACA4421**
- **NACA6412**

Then from the four airfoils we have chosen, we narrow it down by analyzing the performance parameters. Like, among the four we choose the airfoil which has, High stall angle of attack (α_{stall}), suitable Lift-to-Drag ratio $\left(\frac{C_L}{C_D}\right)_{max}$ at cruise, maximum lift coefficient,($C_{L_{max}}$) to be higher than 1.2 and lift coefficient of cruise ($C_{L_{cruise}}$) to reach the value calculated at a decent angle of attack.

6.2.2 Aerodynamic Parameters

We compare plots constructed in XFLR5 software for the selected airfoils at Reynolds' Number of 3.08×10^5 to choose the best one.

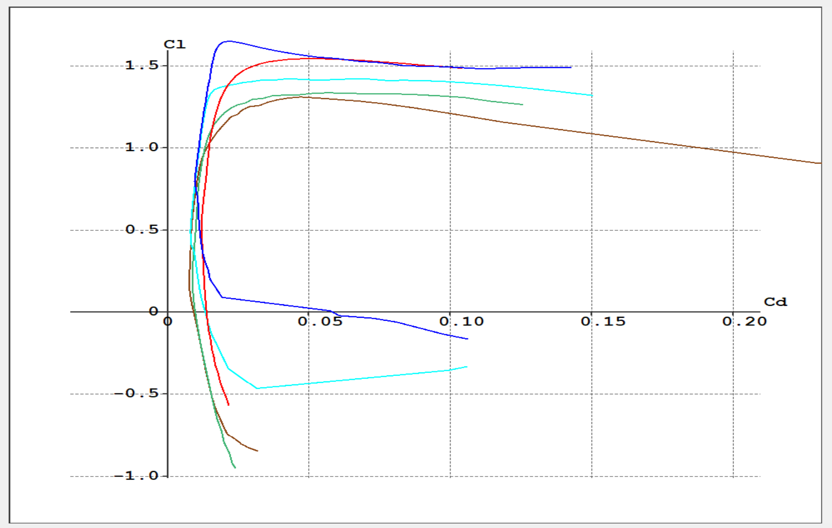

Figure 6.2: C_L vs C_D

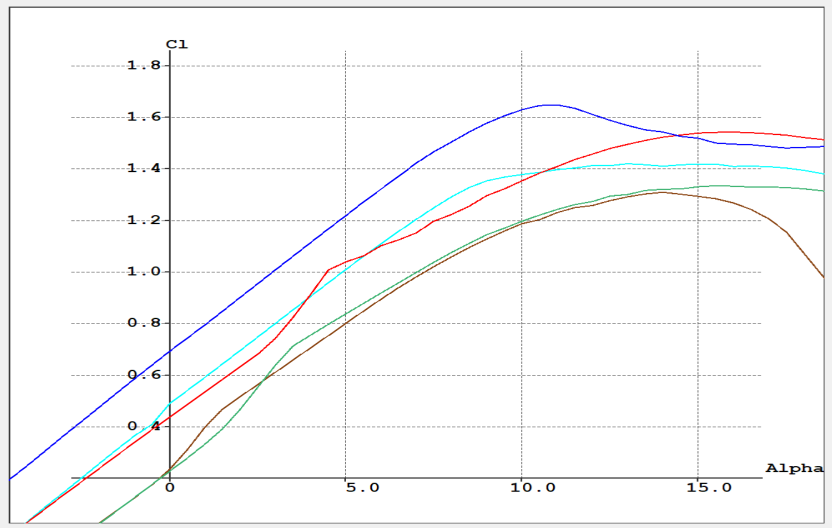

Figure 6.3: C_L vs α

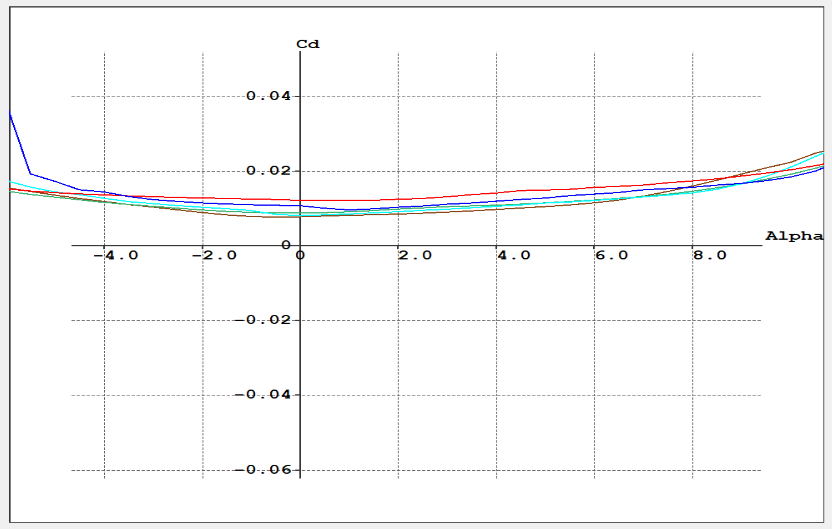

Figure 6.4: C_D vs α

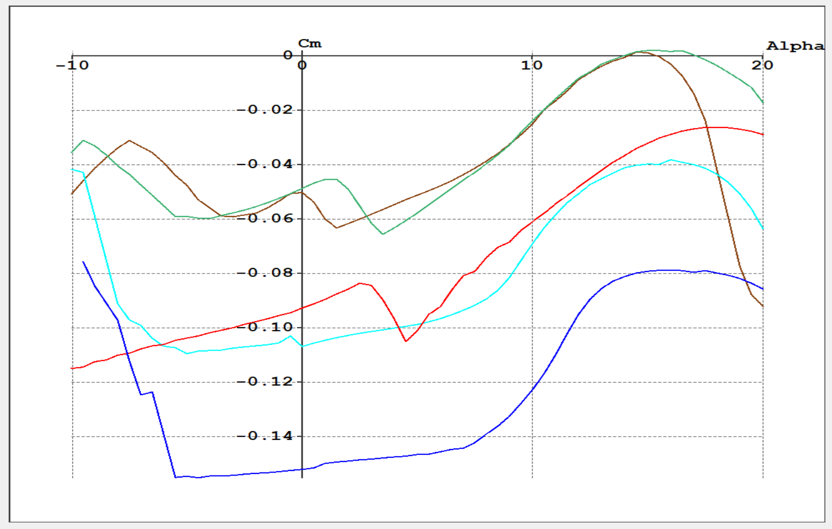

Figure 6.5: C_m vs α

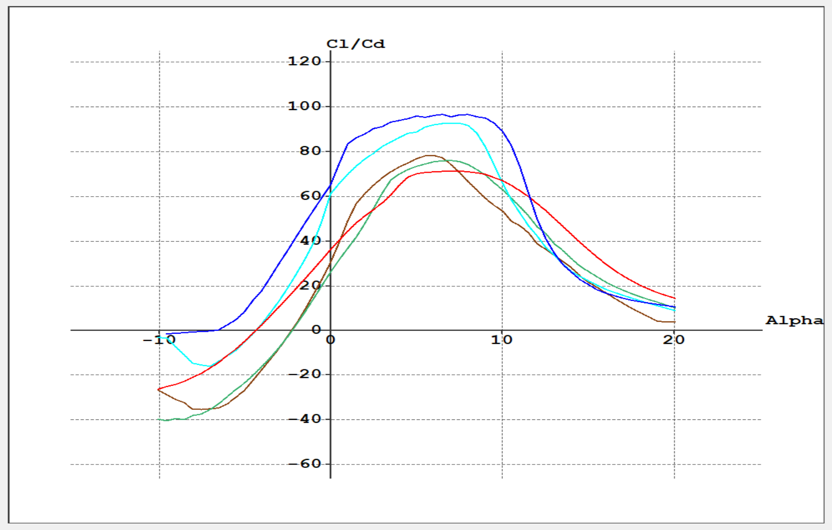

Figure 6.6: $\frac{C_L}{C_D}$ vs α

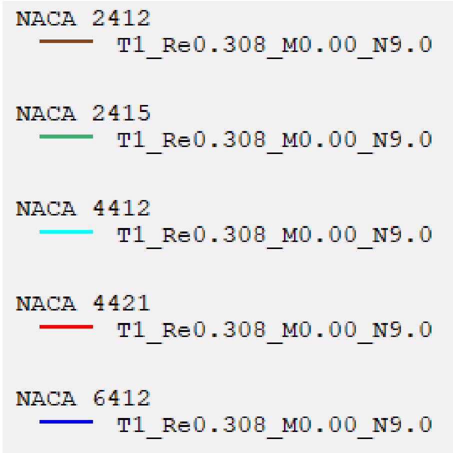

Figure 6.7: Legend for the plots

- We select NACA 2 series airfoil as the $C_{L_{cruise}}$ we require is reached at a decent angle of attack of $3° - 4°$.
- Among NACA2412 and NACA2415, we choose NACA2412 even though both have similar aerodynamic parameters as it is more popular and data is available more readily. It has been proved historically that NACA2412 is very effective.
- C_{L_0} for the airfoil is 0.23.
- C_{L_α} for the airfoil is 0.1195/deg.
- C_{D_0} for the airfoil is 0.008.
- $\alpha_{L=0} = -2.22°$.
- Stall angle around $15 - 17°$.

6.3 Wing Design

6.3.1 Wing Geometry

After choosing the airfoil as NACA2412, a rectangular wing (taper ratio = 1) with AR = 8.0, span, b = 1.84 m and chord, c = 0.231 m with zero twist and zero dihedral angle is chosen. Wing Area = $0.42m^2$ from the DTOW and wing loading value found earlier.

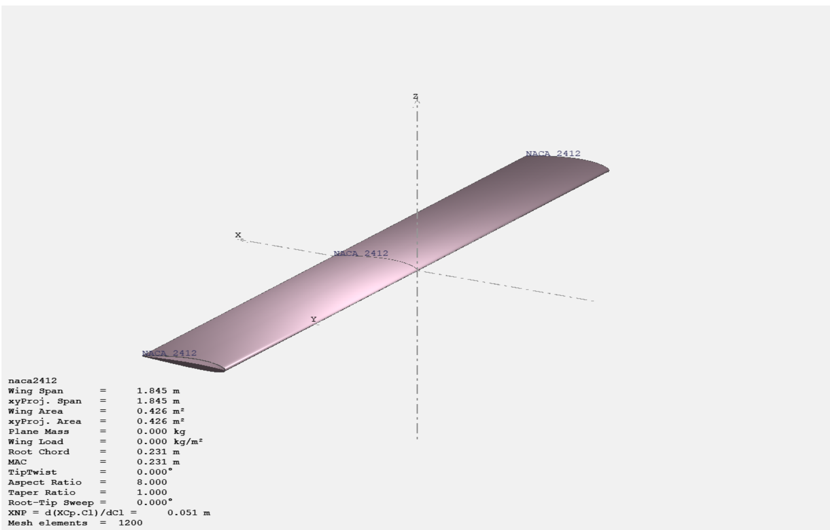

Figure 6.8: Preliminary CAD Model of Wing

Ailerons or flaps are not added as their dimensions are not yet confrimed

6.3.2 Aerodynamic Parameters

Plots made using XFLR5 software showing the aerodynamic characteristics for the wing at Reynolds' Number of 3.08×10^5 is shown below.

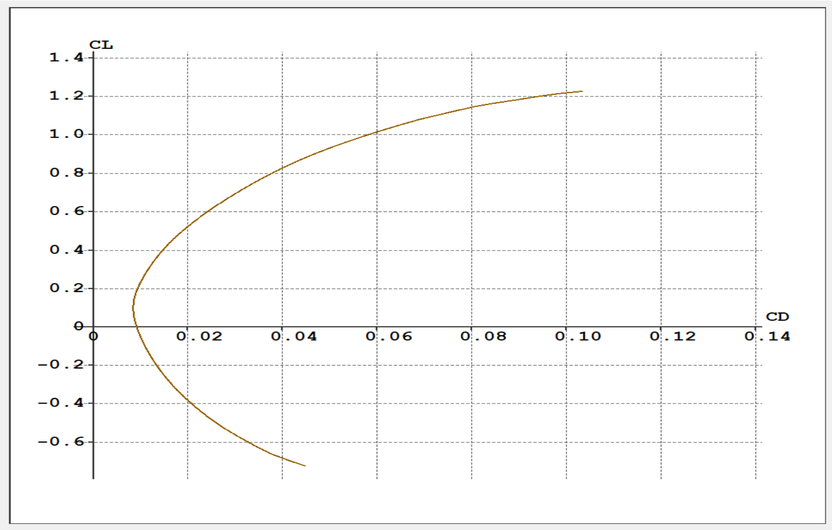

Figure 6.9: C_L vs C_D

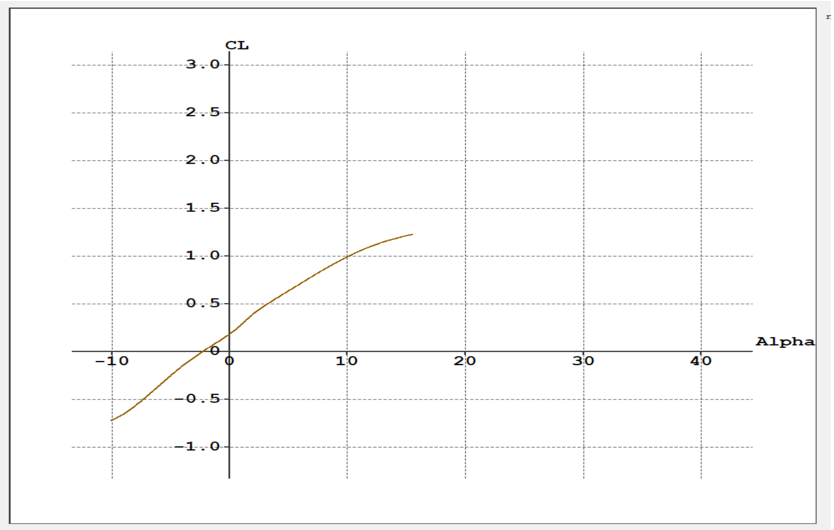

Figure 6.10: C_L vs α

The iterations were not converging for AOA above 15.5. Stall angle can be estimated by a rough extrapolation.

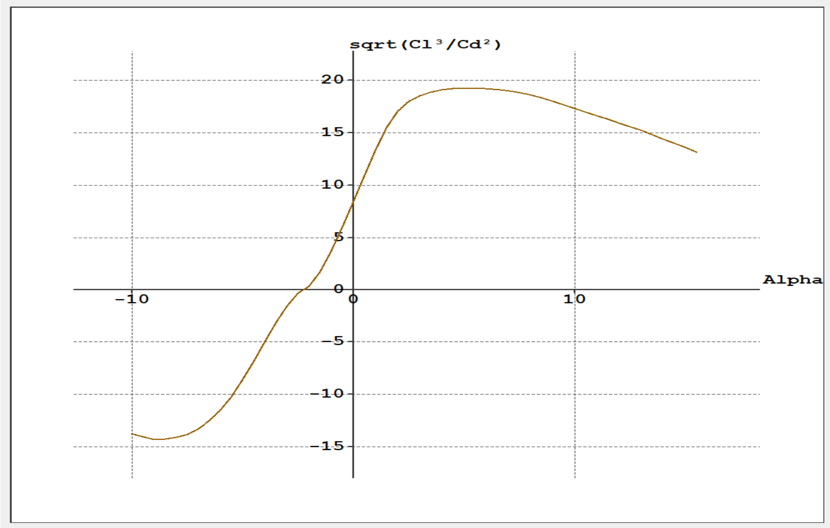

Figure 6.11: $\sqrt{\frac{C_L^3}{C_D^2}}$ vs α

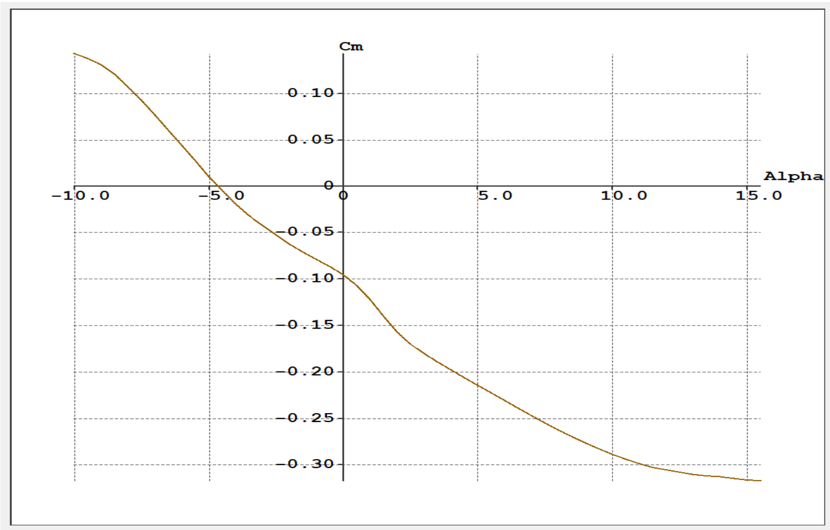

Figure 6.12: C_m vs α

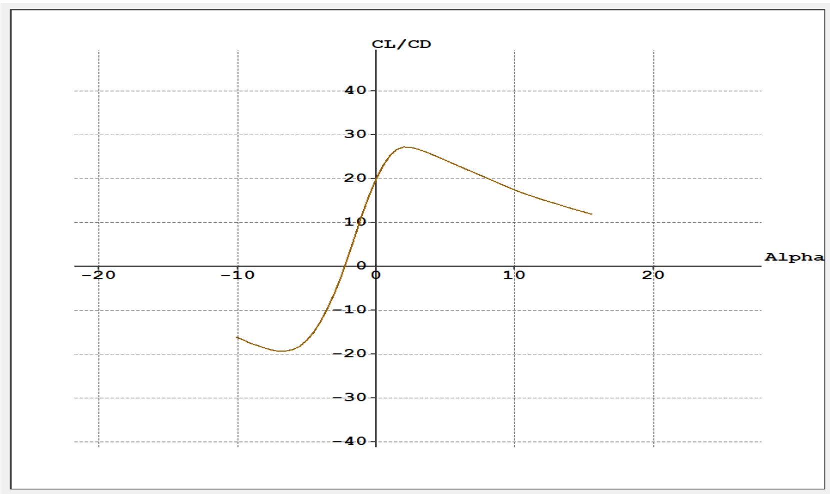

Figure 6.13: $\frac{C_L}{C_D}$ vs α

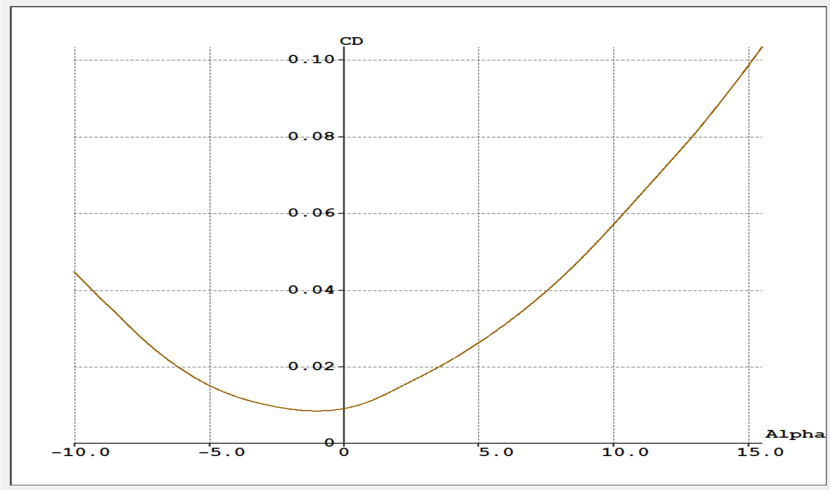

Figure 6.14: C_D vs α

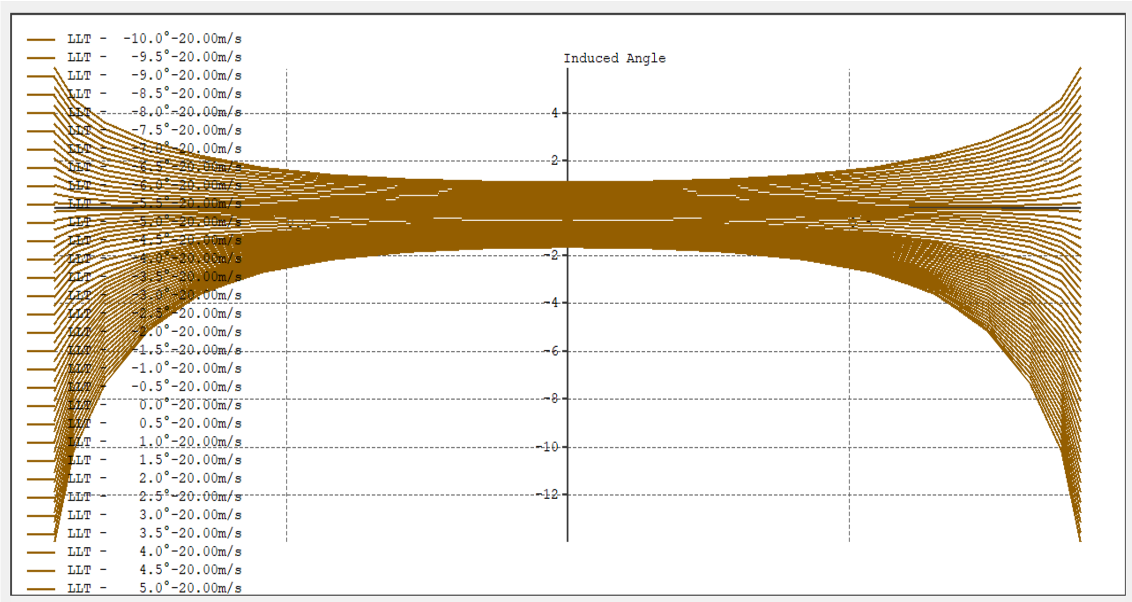

Figure 6.15: Induced angle across span

From the plots, we can infer the following:

- C_{L_0} for the wing is 0.178.
- C_{L_α} for the wing is 0.1188/deg.
- C_{D_0} for the wing is 0.009.
- $\alpha_{L=0} = -2.23°$.
- Stall angle of around 17°.

With the chosen Wing loading of around $127 N/m^2$ and DTOW of 5.47 Kg, we get C_L at cruise to be 0.532.

6.3.3 Wing Dihedral

Wing dihedral is the angle of the wing with respect to the horizontal when seen from the front. Positive (tips higher) dihedral tends to roll the aircraft level whenever it is banked. The resulting rolling moment is approximately proportional to the dihedral angle. If the wing is high-mounted, the air being pushed over the top of the fuselage pushes up on the forward wing, providing an increased dihedral effect. The reverse is true for a low-mounted wing. For the simplicity of design and fabrication, the wing is kept high mounted, without dihedral.

6.3.4 Wing Sweep

It increases wing weight, reduces lift by the cosine of the sweep angle, and makes the ailerons and flaps work poorly. Sweep also makes it more likely that the wingtips will strike the ground in a bad landing. For a low speed airplane, the best sweep is usually zero. The exact wing sweep required to avoid shocks depends upon the selected airfoil, thickness ratio, taper ratio, and of course, the desired flight Mach number. Since the project design is a subsonic UAV, so Wing Sweep is avoided.

6.3.5 Wing Taper Ratio

Wing taper ratio A is the ratio between the tip chord and the centerline root chord. An untwisted rectangular wing has about 7% more drag due to lift than an elliptical wing of the same aspect ratio. When a rectangular wing is tapered, the tip chords become shorter, alleviating the undesired effects of the constant-chord rectangular wing.

6.3.6 Wing type

Aspect ratio is a measure of how long and slender the wing appears when seen from above or below.

Low aspect ratio wings are short and sturdy. These are structurally efficient, have high instantaneous roll rate and low supersonic drag. They tend to be used on fighter aircraft and on very high-speed aircraft.

Moderate aspect ratio are general-purpose wing and are very widely used.

High aspect ratio are long and slender wing. These are more efficient aerodynamically and have less induced drag at subsonic speeds. They tend to be used by high-altitude subsonic aircraft and by high-performance sailplanes.

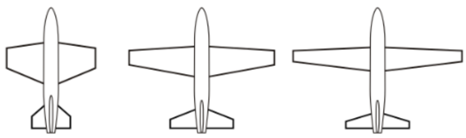

Figure 6.16: Fixed Wing configuration based on Aspect ratio

There are several wing types, but the prominent ones are *Rectangular wing*, *Tapered wing*, *Elliptical wing*, and *Delta wing*.

Rectangular wing

- It is the simplest to manufacture.
- It is a non-tapered, straight wing that is mostly used in small aircrafts. This wing extends out from the aircraft's fuselage at right angles.
- It has larger wingtip vortices and larger downwash towards the tips. Therefore the wing tips have a smaller effective AOA which means they don't generate as much lift.
- Makeflyeasy striver mini binary uav is an example of uav which uses rectangular wing configuration.

Elliptical wing

- It is aerodynamically most efficient because elliptical span-wise lift distribution induces the lowest possible drag.
- However, the manufacturability of this aircraft wing is poor and is quite expensive.
- The elliptical wing wasn't originally designed to minimize drag induction, but rather it was made to house the landing gear along with ammunition and guns inside a wing. So, the wing had to be thin. The ellipse was the shape that allowed for the thinnest possible wing, giving room inside to hold the necessary things.

Figure 6.17: Rectangular wing Aircraft

- In the aircraft like the Seversky P-35, we can see a semi-elliptical wing that has a trailing or leading edge elliptical.
- They produce the same amount of lift across the wing, therefore they have the same effective AOA across the entire span.

Figure 6.18: Elliptical wing Aircraft

Tapered wing

- The wing was designed by modifying the rectangular wing. The chord of the wing is varied across the span for approximate elliptical lift distribution.
- While it isn't as efficient as the standard elliptical wing, it does offer a compromise between efficiency and manufacturability.
- It produces slightly less downwash mid-wing but has quite a significant downwash towards the tip where the tip vortices are quite large and is also difficult to manufacture compared to that of the rectangular wing.

Delta wing

- The delta wing is low aspect ratio wing which is used in supersonic aircrafts.
- The main advantage of a delta wing is that it is efficient in all regimes (supersonic, subsonic, and transonic).

Figure 6.19: Tapered wing Aircraft

- Moreover, this type of wing offers a large area for the shape thereby improving maneuverability and reducing wing loading.
- The delta wing doesn't just offer an efficient flight experience but is also strong structurally and provides a large volume for fuel storage.
- This wing is also simple to manufacture and maintain.
- However, like any other type of aircraft wing, the delta wing also has some disadvantages. Due to their low aspect ratio, delta wings induce high drag.
- At low speed, during landing and takeoff, these wings have a high angle of attack.

Figure 6.20: Delta wing Aircraft

6.3.7 Wing Location

The primary criterion for selecting the wing location comes from the operational requirements, while the other requirements such as stability and manufacturability influence some designs. The final selection is made based on the summations of all advantages and disadvantages when incorporated into the design requirements.
There are four ways for wing placement, each with particular advantages and disadvantages.

High Wing

Advantages:

- Facilitates the installation of an engine on the wing since the engine/propeller clearance is higher (and safer) compared with a low-wing configuration.
- Facilitates the installation of a strut.
- The aircraft structure is lighter when struts are employed.

Figure 6.21: High wing Design

- Increases the dihedral effect. It makes the aircraft laterally more stable. The reason lies in the higher contribution of the fuselage to the wing dihedral effect.
- The wing will produce more lift compared with a mid- and low wing since two parts of the wing are attached at least on the top part.
- The aircraft will have a lower stall speed, since $C_{L_{max}}$ will be higher.

Disadvantages:

- The wing drag produces a nose-up pitching moment, so it is longitudinally destabilizing. This is due to the higher location of the wing drag line relative to the aircraft center of gravity.
- The ground effect is lower, compared with low wing. During take-off and landing operations, the ground will influence the wing pressure distribution. The wing lift will be slightly lower than for the low-wing configuration.
- The landing gear is longer if connected to the wing.
- The horizontal tail area of an aircraft with a high wing is about 20% larger than that of a low wing. This is due to more downwash of a high wing on the tail.
- A high wing is structurally about 20% heavier than a low wing.
- The aircraft lateral control is weaker compared with mid-wing and low-wing since the aircraft has more laterally dynamic stability.

Low Wing

Figure 6.22: Low wing Design

Advantages:

- The aircraft take-off performance is better, compared with a high-wing configuration, due to the ground effect.
- The landing gear is shorter if connected to the wing.
- The aircraft is lighter compared with a high-wing structure.
- The aircraft has higher lateral control compared with a high-wing configuration.
- The wing has less downwash on the tail, so the tail is more effective.
- The wing drag produces a nose-down pitching moment, so a low wing is longitudinally stabilizing. This is due to the lower position of the wing drag line relative to the aircraft's center of gravity.

Disadvantages:

- The aircraft has a lower landing performance since it needs more landing runway.
- The wing generates less lift, compared with a high-wing configuration, since the wing has two separate sections.
- The aircraft will have a higher stall speed compared with a high-wing configuration, due to a lower $C_{L_{max}}$.

Mid Wing

Figure 6.23: Mid wing Design

Mid-wing has the best of both high and low-wing designs. Advantages:

- The mid-wing has less interference drag than the low-wing or high-wing
- The mid-wing is aerodynamically streamlined compared with the two other configurations.
- A strut is usually not used to reinforce the wing structure.

Disadvantages:

- The aircraft structure is heavier, due to the necessity of reinforcing the wing root at the intersection with the fuselage.

- The mid-wing is more expensive and harder to build compared with high- and low-wing configurations.

4) Parasol Wing

Figure 6.24: Parasol wing Design

Parasol wing is generally employed in hang gliders or amphibian aircraft. Several of the features are similar to high wing design.

When selecting a wing location for our UAV, we found that the high-wing design is the best choice due to its sufficient clearance for propellers, easy implementation of strut for structural support, and higher aerodynamic characteristics, including a higher $C_{L_{max}}$.

6.3.8 Ailerons

According to [11], the dimensions of ailerons of the wing can be found from the ranges as shown below:

$$\frac{S_a}{S} = 0.05 - 0.1$$
$$\frac{b_a}{b} = 0.2 - 0.3$$
$$\frac{c_a}{c} = 0.15 - 0.25$$
$$\frac{b_{a_i}}{b} = 0.6 - 0.8$$

The definitions of the terms are given in the image below.

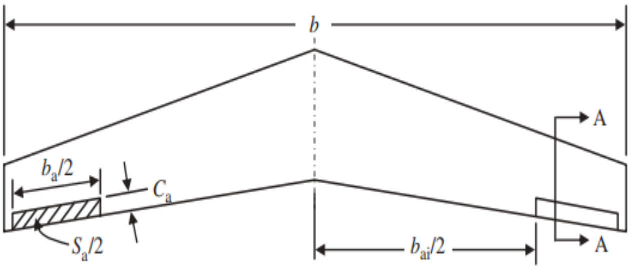

Figure 6.25: Aileron Geometry

Due to lack of further information at this point, we choose the average value of chord and span to find the aileron geometry.

$$b_a = 0.25b = 0.46m$$
$$c_a = 0.2c = 0.046m$$
$$b_{a_i} = 0.7b = 1.288m$$

For S_a:

$$S_a = b_a \times c_a = 0.0212m/s^2$$

6.3.9 High Lift Devices

In addition to the Primary control surface on the wing, the Ailerons, we also have secondary control surfaces known as High lift devices to aid the UAV for certain segments of the mission when there is high lift required at lower speeds. Few of the commonly used High lift devices are Slats, and slots which are located in the Leading edge of the wing and Flaps which are deployed at the Trailing edge of the wing. Slats and slots are primary used in aircraft and not a particular choice when it comes to UAVs because it is difficult to fabricate a wing with slats and slots. For our mission, we will be needing comparatively more lift during climb than during cruise. In order to achieve this, we will be using a Flap at the Trailing edge of the wing.
Generally, while designing flaps, the flap length is taken to be 20%-30% of the Chord length of the airfoil. [10] Since we have finalized our Airfoil to be NACA2412 and wing geometry, A wing is designed with required Geometry in Xflr5 using NACA airfoil as its cross section at different angles of flap deflection and analyzed.

$$\text{Wing Area} = 0.42\ m^2$$
$$\text{Span} = 1.84\ m$$
$$\text{Aspect Ratio} = 8.00$$
$$\text{Chord length} = 0.23\ m$$
$$\text{Flap Length} = 1.00\ m$$

Using these parameters, varying the flap deflection from 10° to 30°, The wing is analysed and the Aerodynamic Characteristics are compared.

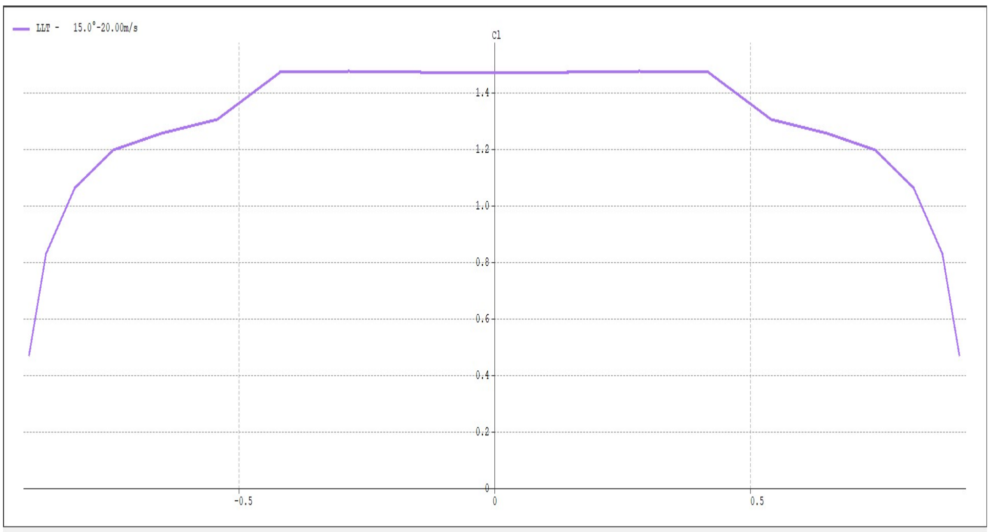

Figure 6.26: Lift Distribution over wing with 10° Deflection

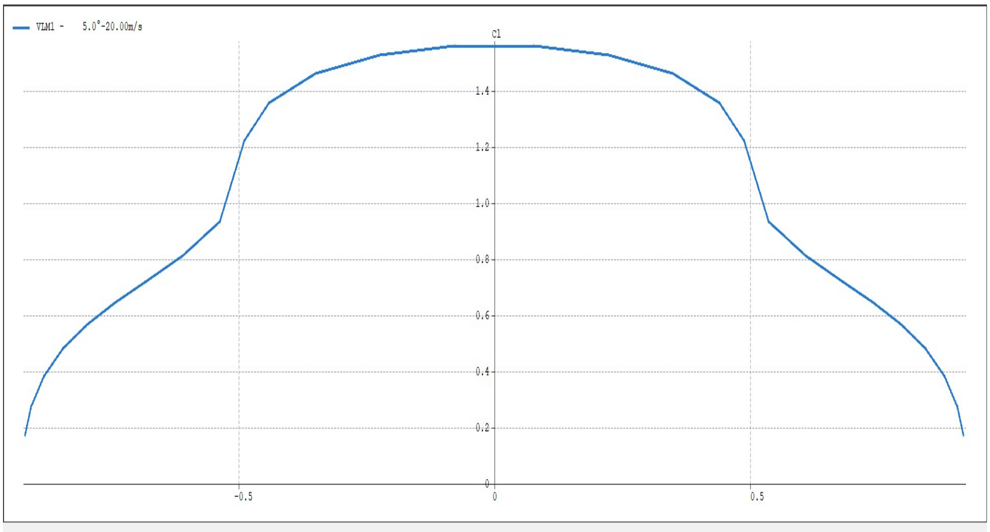

Figure 6.27: Lift Distribution over wing with 20° Deflection

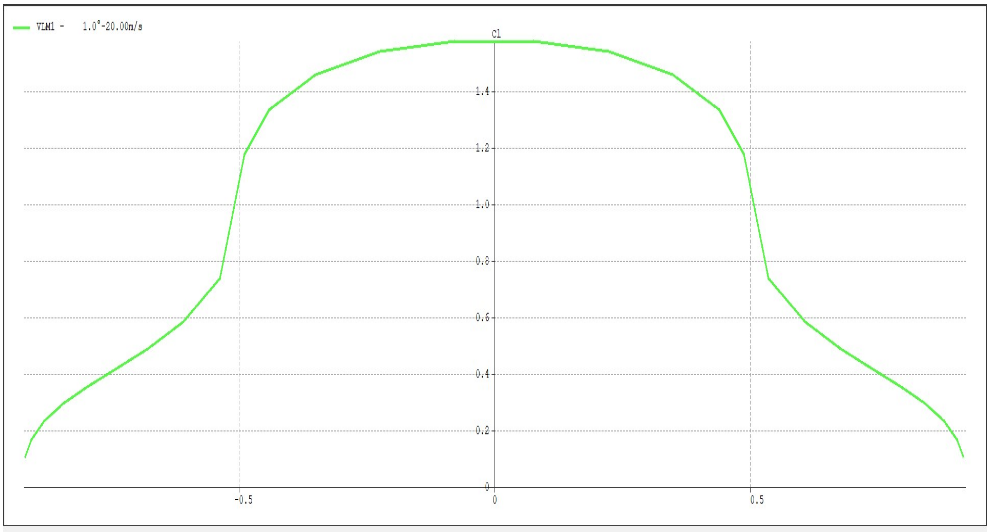

Figure 6.28: Lift Distribution over wing with 30° Deflection

Now Comparing the Aerodynamic Characteristics, Like C_L vs α, $\frac{C_L}{C_D}$ vs α, etc of the wing

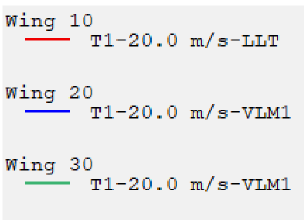

Figure 6.29: Legend for the below plots

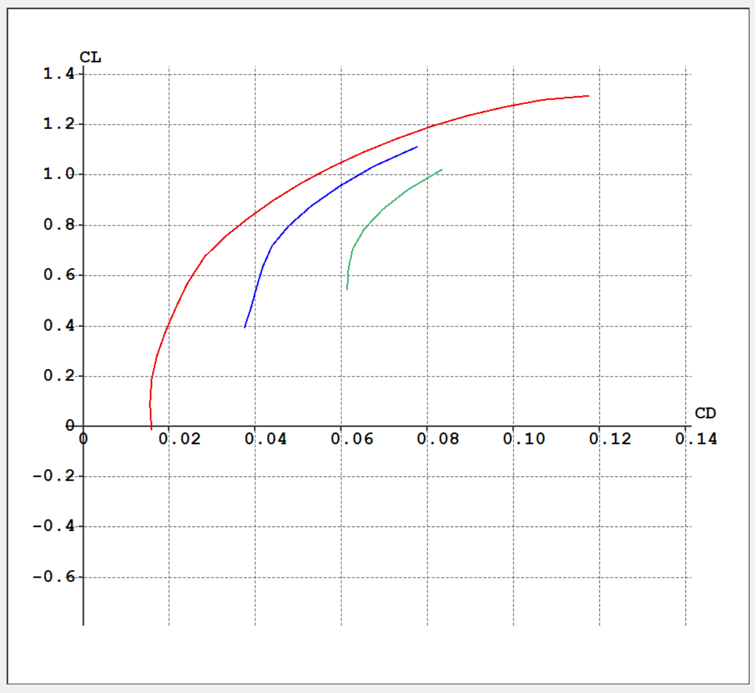

Figure 6.30: C_L vs C_D with flap

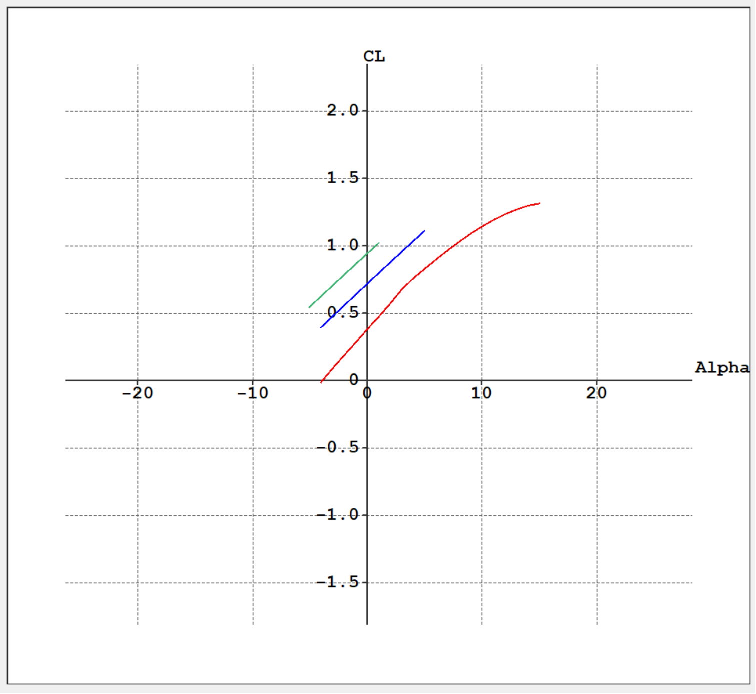

Figure 6.31: C_L vs α with flap

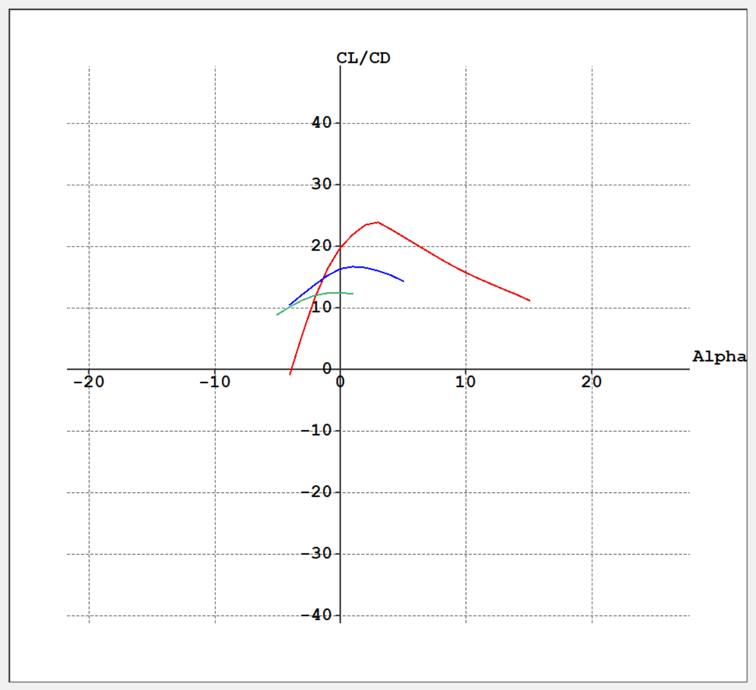

Figure 6.32: $\frac{C_L}{C_D}$ vs α with flap

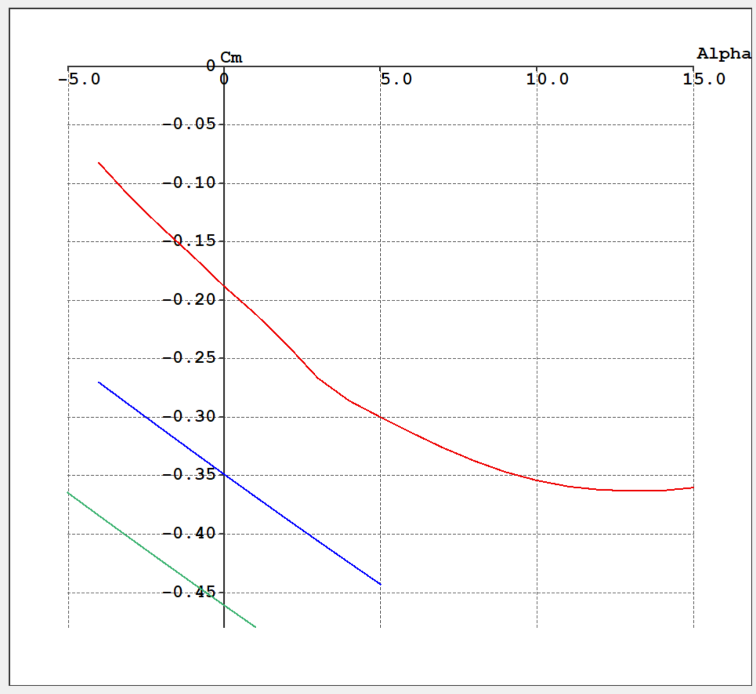

Figure 6.33: C_m vs α with flap

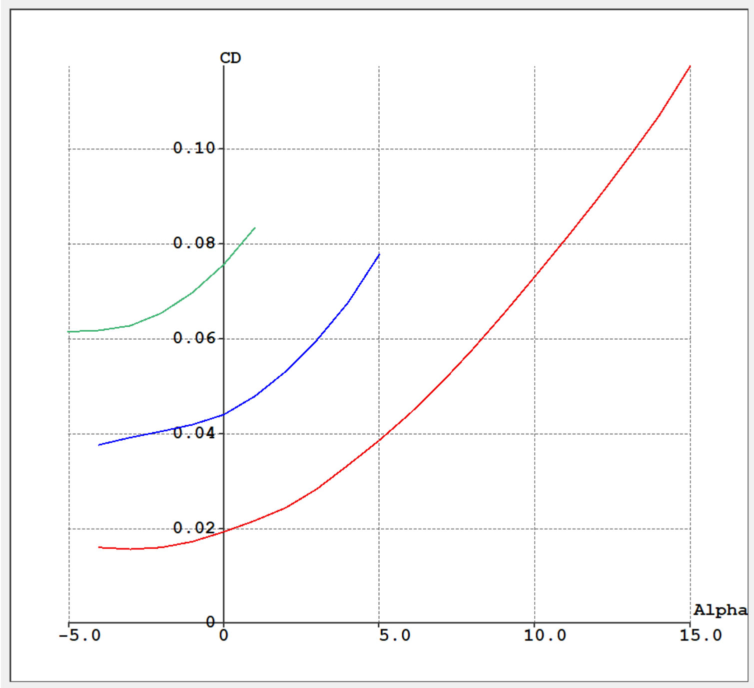

Figure 6.34: C_D vs α with flap

In all 3 cases, we meet the requirement, That is getting a $C_{L_{Max}}$ of 1.2, however the Wing with 10° of flap deflection is able to produce it with minimal parasite drag. In addition to that, Wings with 20° and 30° of flap deflection does not converge to a solution after 5° and 1° Angle of Attack which happens when the wing undergoes stall at that angle of attack. Taking all these factors into acccount, we can limit the Maximum deflection of the Flap at 10°.

Morover, since, flap increases camber and increases effective area of wing, we would require less lift, which definitely helps our cause.

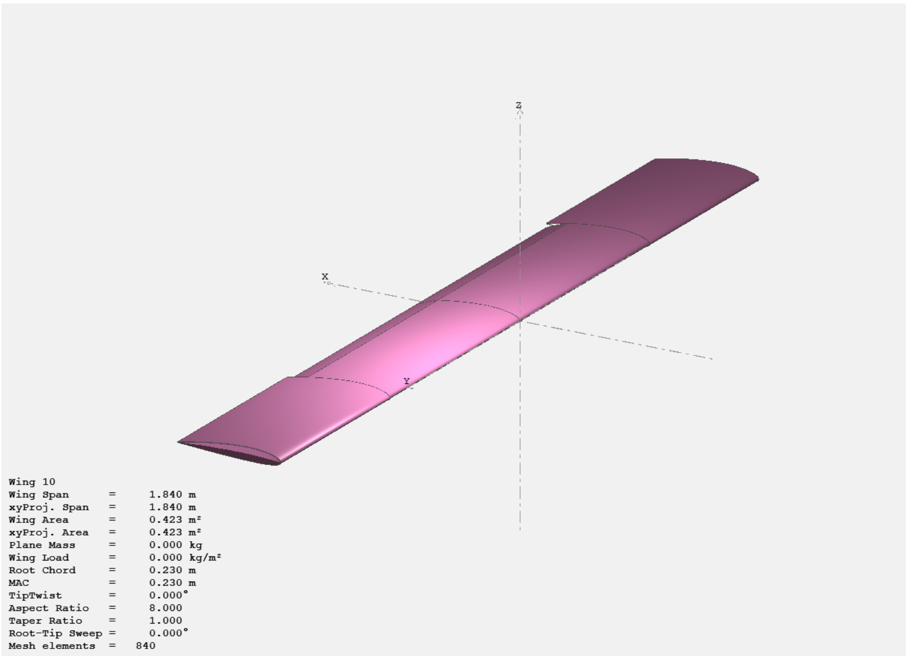

Figure 6.35: CAD model of wing with flap

Lift Requirements will be further studied and flap will be modified accordingly.

6.3.10 Wing Incidence

The angle formed by the extension of chord line and the longitudinal axis is the wing incidence angle.

Figure 6.36: Wing Incidence Angle

$C_{L_{\alpha w}}$ can be found from the C_{L_α} of the airfoil by the relation given in [10],

$$C_{L_{\alpha w}} = \frac{C_{L_\alpha}}{\sqrt{1 - M_\infty^2} + \frac{C_{L_\alpha}}{\pi e AR}}$$

We can neglect the square of mach number. Substituting values, we get $C_{L_{\alpha w}} = 0.1188/deg$ Flaps will not change the $C_{L_{alpha_w}}$ at moderate angles of attack. It however changes the $\alpha_{L=0}$ to $-3.866°$.

Since we are not considering any twist in the wing, and using $\alpha_{FRL} = 0$ the incidence angle can be found from the relation :

$$C_L = C_{L_\alpha}(\alpha_{FRL} + i_w - \alpha_{L=0})$$
$$0.532 = 0.1188(i_w + 3.866)$$

From this, we get the wing incidence angle, $i_w = 0.612°$.

Chapter 7

Tail and Fuselage Design

7.1 Tail

7.1.1 Tail Configuration

The conventional tail configuration is chosen as it is the simplest configuration with and most convenient to perform all tail functions (i.e, trim stability and control). This configuration includes one horizontal tail (two left and right sections) located on the aft fuselage, and one vertical tail (one section) located on top of the aft fuselage. Both horizontal and vertical tails are located and mounted to the aft of the fuselage. The horizontal tail is mainly employed to satisfy the longitudinal trim and stability requirements, while the vertical tail is mainly used to satisfy the directional trim and stability requirements. Taking into account, the challenges faced due to lack of experience and cost during fabrication, the conventional tail configuration is the best choice.

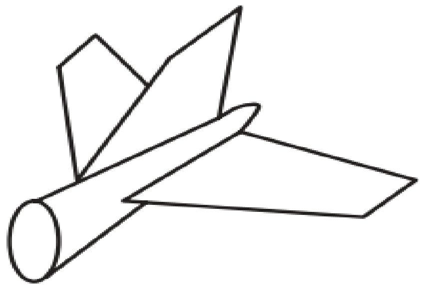

Figure 7.1: Conventional Tail Configuration

7.1.2 Data Collection For Tail

The values of l_h are found from image processing using Fusion 360.

UAV	$\mathbf{l_h}$	HTVR
Raven B	0.667	0.5364
Puma LE	1.577	0.6565
Elbit Skylark I	1.321	0.8601
Horus FT100	0.85	0.6876

Some Notations

- l_h is the horizontal tail moment arm.
- HTVR is the Horizontal Tail Volume Ratio.
- S_h is the horizontal tail planform area.
- l_v is the vertical tail moment arm.
- VTVR is the Vertical Tail Volume Ratio.
- S_v is the vertical tail planform area.

- $\bar{c}$ is the chord of the wing.
- b is the wingspan.
- c_h adn b_h are the horizontal tail chord and span respectively.
- S is the wing planform area.

7.1.3 Horizontal Tail

Configuration

For the same arguments that we put forward for selection of wing configuration, i.e, for the ease of fabrication, a rectangular configuration is chosen for the horizontal tail.

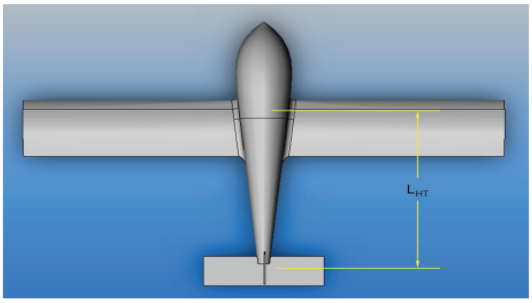

Figure 7.2: Rectangular Horizontal Tail Configuration

Airfoil Selection

A horizontal tailplane is a lifting surface (similar to the wing) and requires a special airfoil section. [11] mentions that the the tailplane airfoil lift curve slope, C_{L_α} must be as large as possible along with a considerably wide usable angle of attack. Since the aircraft center of gravity moves during cruising flight, the airfoil section must be able to create sometimes a positive ($+L_h$) and sometimes a negative lift (-L_h). For this reason, a symmetric airfoil section is a suitable candidate for a horizontal tail. We choose NACA0012 as the airfoil for our aircraft. It is desired that the horizontal tail never stalls, and the wing must stall before the tail.

Parameters for Horizontal Tail

- **Aspect Ratio (AR_{ht})** - [11] gives the relation between AR_t and AR_{wet} :

$$AR_{ht} = \frac{2}{3} AR_w \tag{7.1}$$

  With our Wing Aspect Ratio of 8, we get $AR_{ht} = 5.33$ This is an initial estimate of the horizontal tail aspect ratio. The final value for the tail aspect ratio will be determined based on the aircraft stability and control, cost, and performance analysis evaluations after the other aircraft components have been designed.

- **Taper Ratio (λ_{ht})** - It is defined as the ratio between the tail tip chord and the tail root chord. Again, as per fabrication constraints, we keep taper ratio as 1 for horizontal tail.

- **Sweep Angle and Dihedral** - As an initial estimate we can keep sweep angle and dihedral the same as that of the wing ([11]). The final values will be decided after detailed stability, cost and manufacturing considerations.

- **Horizontal Tail Volume Ratio (HTVR)**: HTVR is defined as follows,

$$HTVR = \overline{V}_H = \frac{l_h}{\bar{c}} \frac{S_h}{S} \tag{7.2}$$

  From the previous aircraft data collected, we get the average of HTVR as 0.685 and we take l_h as 96.3 cm. Using these values and $S = 0.42m^2$ and $\bar{c} = 0.23m$ from wing design, we get

$$S_h = \overline{V}_H S \frac{\bar{c}}{l_h} = 0.069m^2 \tag{7.3}$$

  We can write $S_h = b_h c_h = 0.069m^2$ and $AR_h = \frac{b_h}{c_h} = 5.33$. From these we get $b_h = 0.606m$ and $c_h = 0.1137m$.

Elevator

[11] mentions the following preliminary approximations.

- $\frac{S_E}{S_h}$ = 0.15 - 0.4.
- $\frac{b_E}{b_h}$ = 0.8 - 1.
- $\frac{C_E}{C_h}$ = 0.2 - 0.4.

We can keep these on hold and confirm after stability analysis.

7.1.4 Vertical Tail

Surface Area of Vertical Tail

A standard value of VTVR of 0.05 is taken for our UAV from [11]. $l_{ht} \approx l_v = 96.3$ cm.

$$V_{H_{vt}} = \frac{S_{vt} l_{vt}}{S_w b_w} \tag{7.4}$$

$$S_{vt} = V_{H_{vt}} \times \frac{S_w b_w}{l_{vt}}$$

Substituting the Values in the above Equation, we get,

$$S_{vt} = 0.04 m^{)} 2$$

Taper Ratio of Vertical Tail

As mentioned in [11], taper ratio of 0.4 to 0.6 are typical taper ratios used for vertical tail. Although a taper ratio of 1 might be easy to manufacture, that will result in production of lot of drag compared to the ones which fall in this range. Hence for our Aircraft, we will take the taper ratio as 0.5

$$\lambda_{vt} = 0.5 \tag{7.5}$$

Sweep Angle

We want a simple model for manufacturing feasibility. Hence in our case, we have :

$$\Lambda_{vt} = 0^\circ \tag{7.6}$$

Airfoil Selection

Generally the airfoil used in the tail of the aircraft will be symmetric and for low subsonic Aircrafts, NACA0012 is used. Hence we will be going with NACA0012 as our airfoil for the vertical tail.

Incident Angle

For propeller driven aircrafts, the slipstream of propellers experience swirl which creates asymmetric flow field that depends on the size of the propeller. Since we are in the early stages of design, there is no method to calculate this flow field. Hence at this point, we will assume the incident angle as 0°. This might change in the later parts of the analysis.

Dimensions of Vertical Tail

From $S_v = 0.04m^2$ and AR = 3(since, vertical tail is kind of a half wing),

$$S_v = \frac{1}{2} b_v (c_r + c_t) = 0.04$$

where, c_r and c_t are the root and tip chords respectively. $\lambda_v = 0.5$ gives us,

$$\frac{c_t}{c_r} = 0.5$$

These give us,

- $b_v = 34.64cm$, $c_r = 15.4cm$, $c_t = 7.7cm$

7.1.5 Rudder

[11] mentions the following preliminary approximations.

- $\frac{S_r}{S_v} = 0.15$ - 0.35.
- $\frac{b_r}{b_v} = 0.7$ - 1.
- $\frac{c_r}{c_v} = 0.15$ - 0.4.

We will confirm these after stability analysis.

7.2 CAD Model for wing and tail

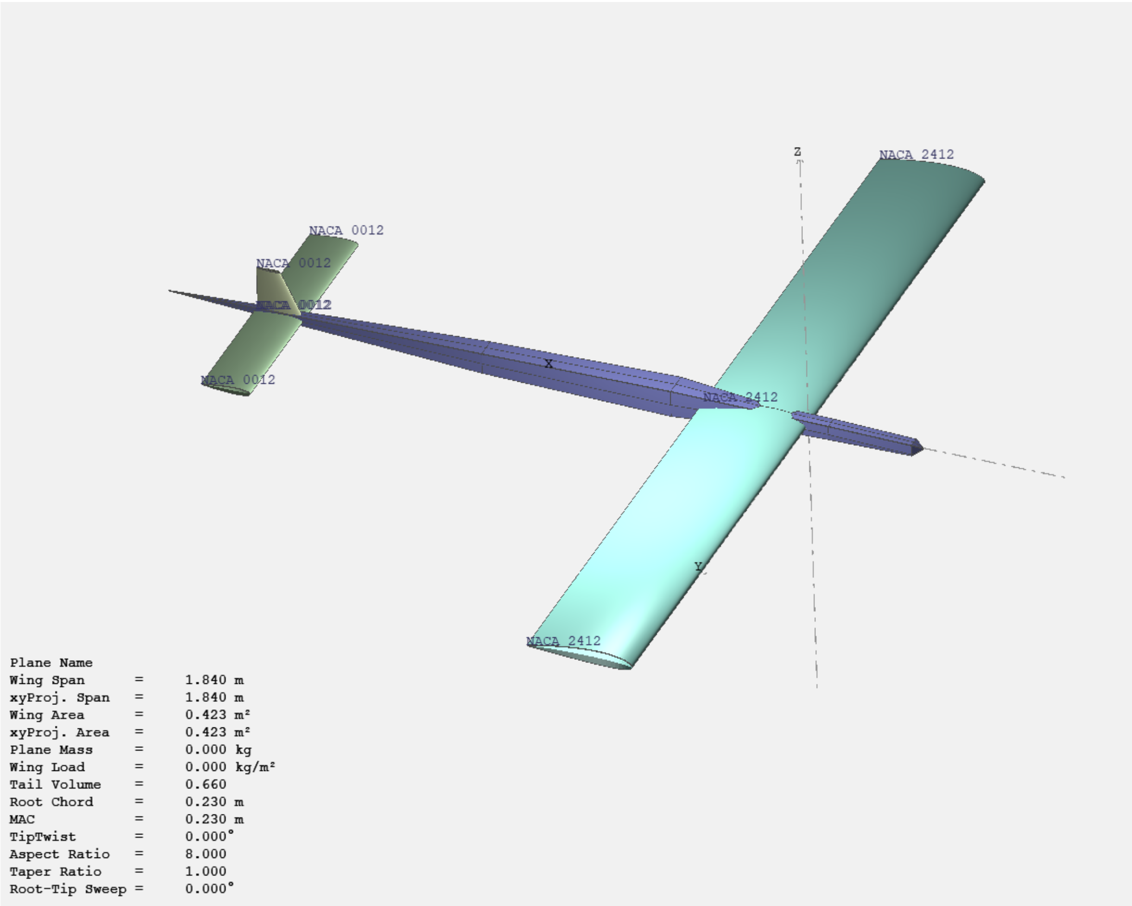

Figure 7.3: CAD model showing preliminary dimensions of tail and wing

These parameters will be finalised after stability analysis.

7.3 Fuselage

7.3.1 Introduction

Fuselage is one of most important aircraft component as its primary function is to accommodate the payload. It has certain design features and expectations, those are:-

1. To have low weight.
2. Generate the lowest drag.
3. Contribute positively to the lift generation.
4. Low wetted area etc.

The fuselage drag usually contributes 30–50% of the aircraft zero-lift drag (C_{D_o}). Furthermore, the fuselage may be aerodynamically designed such that it provides as much as 50% of the total lift.

The fuselage length (L_f) and maximum diameter (D_f) are the two main characteristics that need to be established throughout the design phase. These two dimensions and the fuselage configuration are products of many design requirements. The general requirements for fuselage design are as follows:

1. operational and mission requirements.
2. aerodynamic requirement and Stability requirement.
3. low weight, low wetted area and low side area.
4. symmetry and structural integrity.
5. manufacturability and cost etc...

7.3.2 Fuselage Length(L_f)

[10] gives a relation to obtain an initial value of fuselage length (L) from the MTOW (W_0) of the aircraft.

$$L = aW_0^c \tag{7.7}$$

UAV	**MTOW ($\mathbf{W_0}$) (Kg)**	**Length (m)**
Raven B	2.2	1.1
Puma LE	10.7	2.2
Elbit Skylark I	7	2.25
Horus FT100	8	1.95
Sirius Pro	1.7	1.25
Mini Shark	5.5	1.3
Desert Hawk III	4	1.11

We use previous aircraft data to obtain the value of the constants a and c by curve fitting and subsequently use the relation for out DTOW of 5.47 Kg to obtain the initial value of fusleage length. Refer Appendix A.4 for the code used.

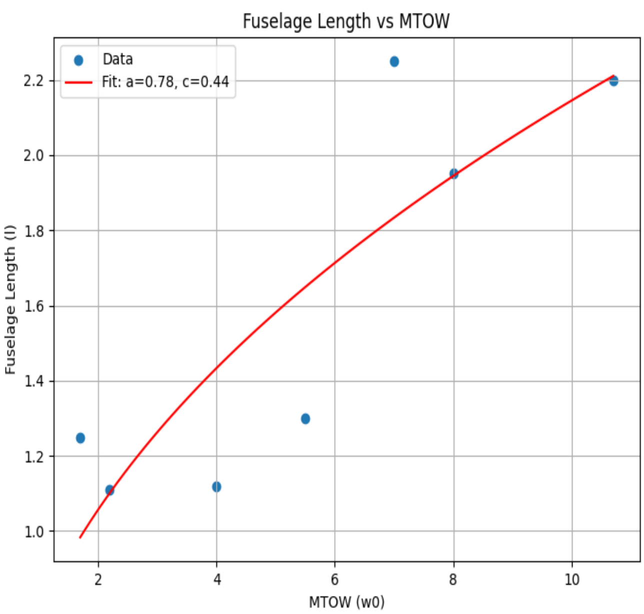

Figure 7.4: Fuselage Length vs MTOW

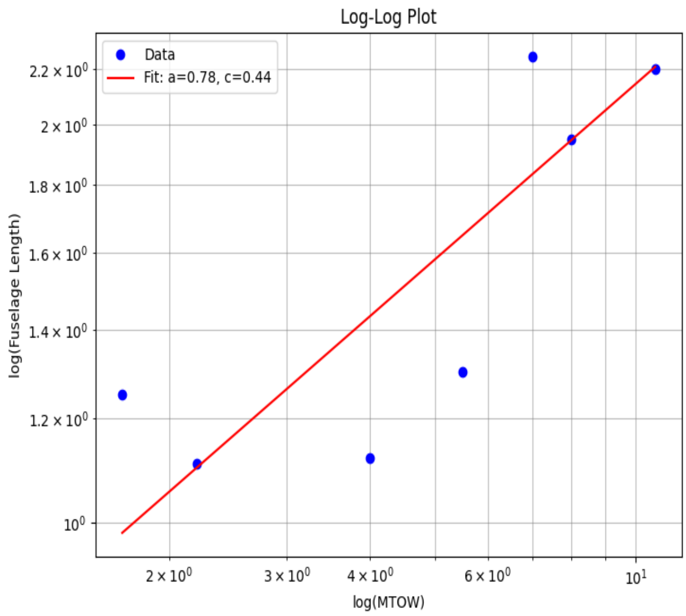

Figure 7.5: log-log plot

From the log-log plot, we get the values of a = 0.78 and c = 0.44. Calculating fuselage length,

$$L = 0.78 \times (5.47^{0.44}) = 1.65m \tag{7.8}$$

We take L = 1.69m as our fuselage length.

7.3.3 Fuselage Width and Height

We will use previous aircraft data to find a linear relation between fuselage length and diameter as well as fuselage length and height. The Fuselage height and width are found from image processing in Fusion 360.

UAV	**Length (m)**	**Width (m)**	**Height** (m)
Raven B	0.9	0.13	0.095
Puma LE	2.1	0.27	0.084
Elbit Skylark I	2.2	0.15	0.16
Horus FT100	1.9	0.14	0.22

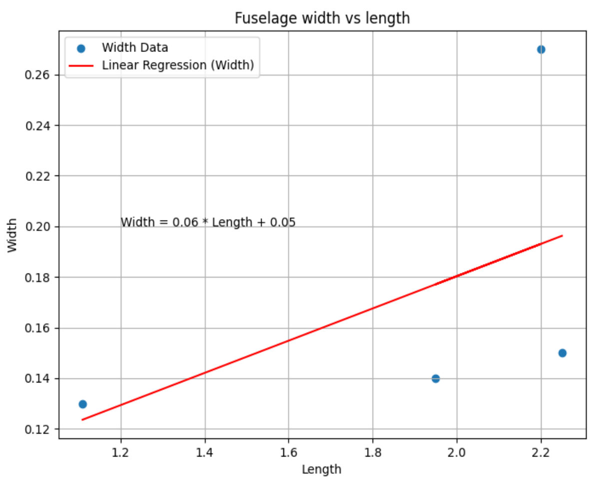

Figure 7.6: Using Linear Regression to find width of fuselage

For a Length of 1.69m, we find the width to be 16.81cm.

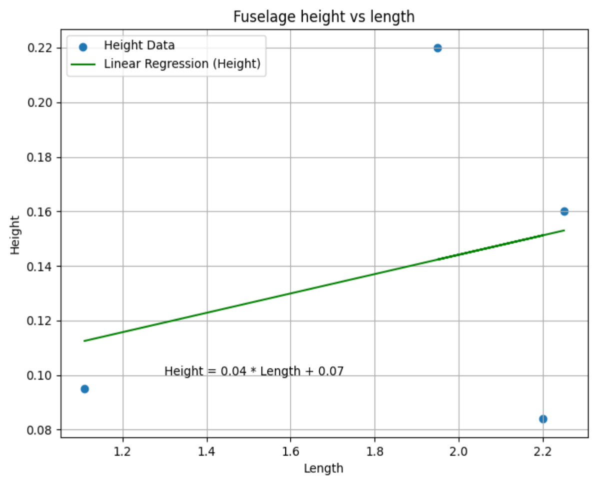

Figure 7.7: Using Linear Regression to find height of fuselage

For a Length of 1.69m, we find the height to be 13.66 cm.
Refer Apendix A.5 for the script used.

7.3.4 Fuselage layout

Determining the fuselage configuration and interior organization comes first in the design process, following the identification of the cargo and design requirements. A conceptual design at the fuselage level, fuselage configuration design does not require extensive calculation. This will give us the rough idea of placement of our internal components, payload etc... The exact locations can be determined during stability analysis such that CG placement can be under our control. The figure is our rough layout estimation.Battery and electronics are the only components to be kept in the fuselage. These are placed in fuselage such that we get CG at the desired location.

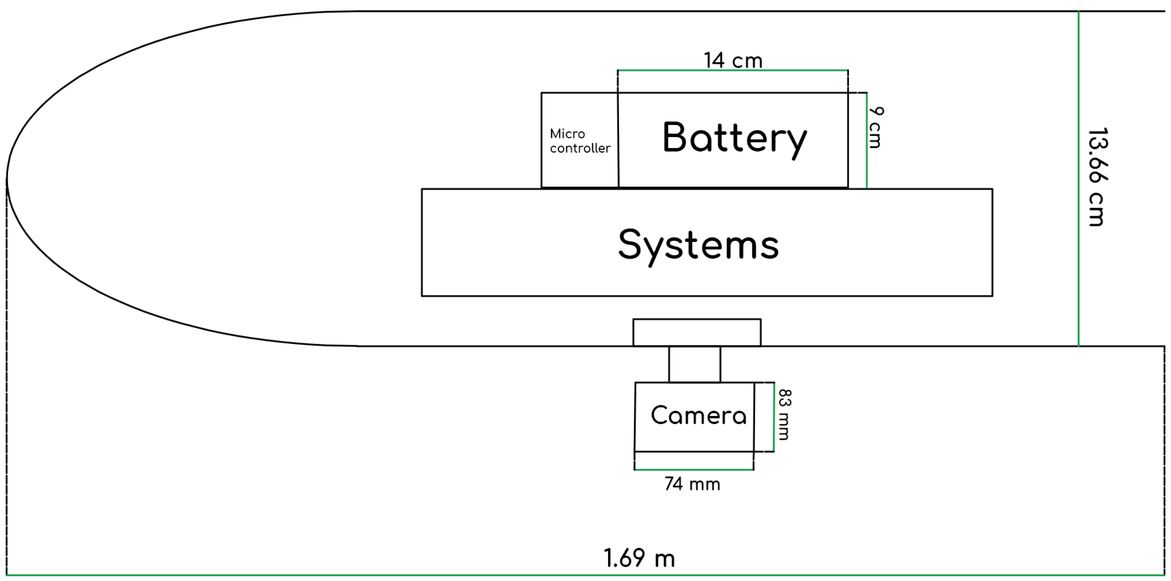

Figure 7.8: Primary layout of UAV

Battery Dimensions : 14 x 10 x 9(cm)

Chapter 8

Landing Gear

8.1 Configuration

No.		Single main	Bicycle	Tail-gear	Nose-gear	Quadricycle	Multi-bogey	Human leg
1	Cost	9	7	6	4	2	1	10
2	Aircraft weight	3	4	6	7	9	10	1
3	manufacturability	3	4	5	7	9	1	10
4	Take-off/landing run	3	4	6	10	5	8	2
5	Stability on the ground	1	2	7	9	10	8	5
6	Stability during taxi	2	3	1	8	10	9	–

10: best, 1: worst.

Figure 8.1: Table for comparing different configurations, fro [11]

The preferred choice is the tricycle landing gear configuration. Some features are:

- Tricycle is the most widely used landing gear configuration.
- The wheels aft of the aircraft CG are very close to it compared to the forward gear and carry 80-90% of the aircraft weight.
- Both main and nose gears have the same height, so the aircraft is level on the ground, although the main gears tend to have larger wheels.
- A nose-gear configuration aircraft is directionally stable on the ground as well as during taxiing.

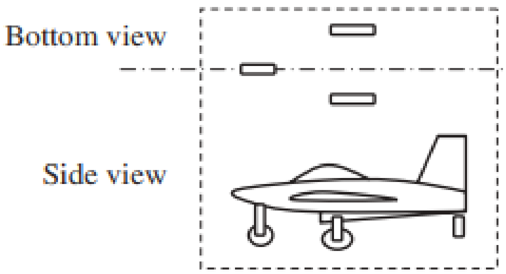

Figure 8.2: Tricycle Landing gear configuration, taken from [11]

8.2 Notations

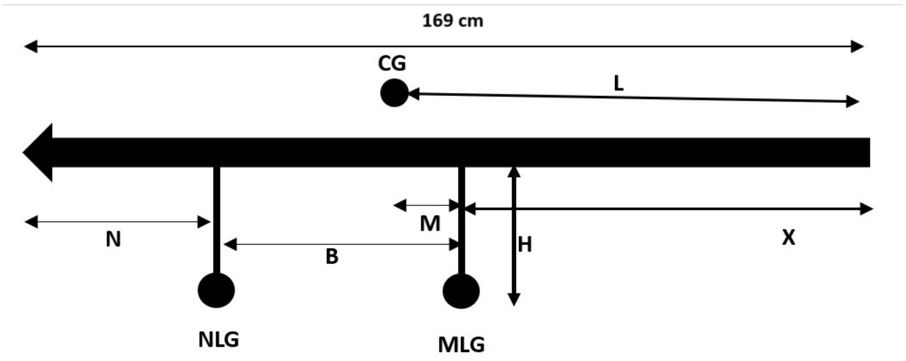

Figure 8.3: Rough Sketch of Landing Gear Position

From tail and fuselage design, we found the Fuselage length = 169cm. We are assuming the main landing gear to carry 85% load and the nose landing gear to carry the remaining 15%load as given in [11].

The other parameters are as follows:

- Location of CG from the aft end = L
- Landing gear height = H
- Wheel Base = B
- Distance of Main Landing Gear (MLG) from tail = X
- Distance of Nose Landing Gear (NLG) from Nose = N
- Distance between CG and MLG = M
- Take off attitude of aircraft = α

As an initial estimate we take L = 1.2 m from the aft end to be the location of CG, which is slighly higher than the Tail moment arm.

Taking moment about CG,

$$0.15W \times N = 0.85W \times M$$
$$N + M = B$$

We get, distance between CG and MLG =$M = 0.15B$.
After estimating Wheel base, B, we can use these relations to estimate the values of the parameters. We can later use these values for stability analysis to finalise them.

8.3 Landing Gear Height

Landing gear height is estimated so that none of the components strike the surface while the aircraft is taxing. The components which can strike the ground during ground roll are the camera and the propeller. We are using a 15 inch diameter propeller. propeller is attached on the wing.

- Height of fuselage = 13.66 cm.
- Height of camera = 13.9 cm.
- Radius of propeller = 18.9 cm.

So, propeller reaches 18.9 - 13.66 = 5.24 cm below the fuselage. So, the camera is the longest component below the fuselage extending to about ¡camera height ¿cm. We take the landing gear height to be 20cm which gives enough clearance for the camera and propeller.

$$H = 21.56cm$$

8.4 Main Landing gear position

[11] mentions that 10–15 deg, so the tipback angle must be equal to or greater than 15–20 °. We take that angle as 16°. With this and landing gear height, $H = 21.56cm$ and fuselage height of 13.66 cm, we get distance from tail to position of main landing gear, $X = 99cm$.

8.5 Ground Clearance and Takeoff Angle

[11] mentions that the takeoff angle, α_{TO} should be 5° less than the tipback angle. Hence, we get $\alpha_{TO} = 11°$.

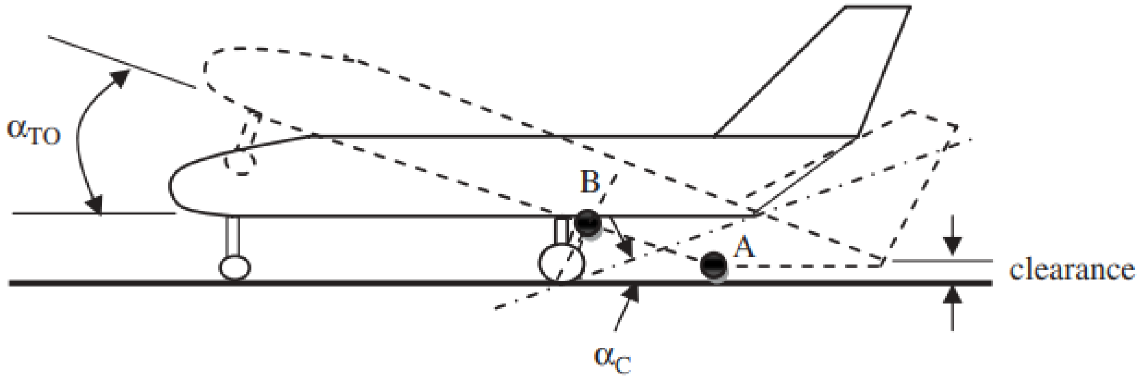

Figure 8.4: Ground clearance, from [11]

8.6 Wheel Base

Wheel base (B) plays an important role in the load distribution between primary (i.e., main) gear and secondary (e.g., nose or tail) gear. This parameter also influences the ground controllability and ground stability.Due to the ground controllability requirement, the nose gear must not carry less than about 5% of the total load and also must not carry more than about 20% of the total load (e.g., aircraft weight). Thus, the main gear carries about 80–95% of the aircraft load. To meet this requirement, it is decided that the nose gear should carry 15% of the total load and the main gear 85% of the total load. [11]

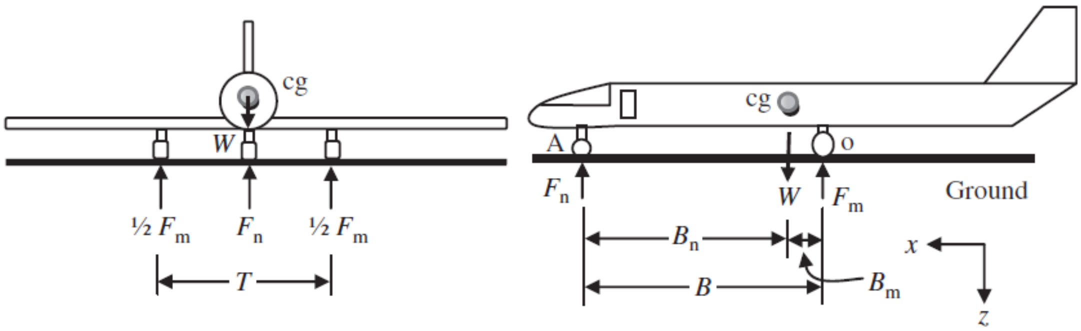

Figure 8.5: Wheel Base

The nose landing gear would be kept where the nose ends and fuselage starts, which is kept at 15cm, from the nose end. Since, $X = 99cm$ and total length of plane, $L = 1.69m$ we get the wheel base, B as ,

$$L = B + 99cm + 15cm$$
$$B = 55cm$$

Therefore,

1. Wheel Base (B) = 55 cm.
2. Distance of nose gear from CG (B_n) = 0.81 B = 44.55 cm.
3. Distance of main gear from CG (B_m) = 0.19 B = 10.45 cm.

8.7 Wheel Track

Wheel Track (T) is the distance between the right and leftmost wheels when looking at the front view. Three main requirements which drive the magnitude of wheel track are : 1) Ground lateral control, 2) Ground lateral stability, and 3) Structural integrity.
To determine the wheel track, the overturn angle is introduced. When looking at the front view the angles between the vertical line passing through the cg and line between aircraft cg and main wheel is known as overturn angle.

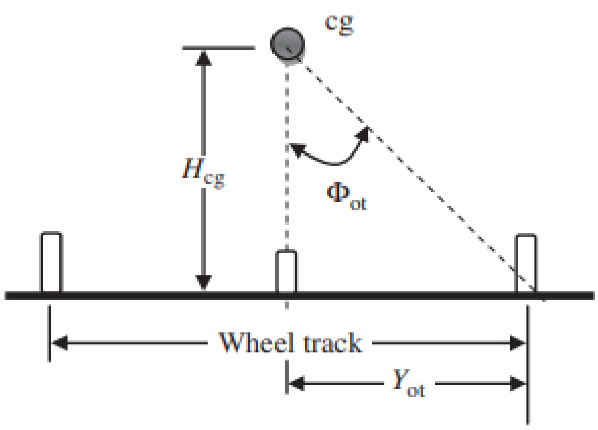

Figure 8.6: Overturn Angle
[11]

As informed in [11] the overturn angle ϕ_{ot} is inside the following limit:

$$\phi_{ot} \geq 25^o$$

As seen from the figure,

$$\tan \phi_{ot} = \frac{Y_{ot}}{H_{cg}}$$
$$Y_{ot} \geq H_{cg} \tan 25^o$$
$$Y_{ot} \geq (14.73 + 0.5 \times 13.66)(0.46630)$$
$$Y_{ot} \geq 10.05\ cm$$

Also,

$$T = 2Y_{ot}$$

Hence,

$$T \geq 20.1\ cm$$

We keep the wheel track, $T = 20.1$ cm for now.
The overturn angle and wheel track can be calculated more precisely after detailed design & sizing.

8.8 Struts

For wheel strut sizing, we will calculate the Cross sectional area for the landing gear strut and the length. Typically cross sections of a landing gear are circular or rectangular in geometry. But since we are going with a non-retractable landing gear, we will be going with a symmetric airfoil to ensure that drag produced is less and streamlined flow is generated. The cross section area of the landing gear strut will be a function of UAV mass, its weight distribution, and the load during touchdown.
Our UAV weighs 6.51 Kg now approximately. We can assume the weight distribution of 15% over the nose landing gear and 85% over the main landing gear. During soft landing, the load factor usually would be close to 1g. Although for cases where there is a need for hard landing, it can go a little over 2g.

$$W_{Nose} = \frac{19}{100} \times 5.47 \times 9.81$$
$$W_{Nose} = 10.19N$$

Load on each of the main landing gear would be

$$W_{main} = 0.5 \times \frac{81}{100} \times 5.47 \times 9.81$$
$$W_{main} = 21.68N$$

where, W_{Nose} and W_{Main} are the loads on nose and main landing gears respectively.
Now, the airfoils used for the landing gear struts are usually Eppler series airfoil which are thick so it can withstand more axial stress. one of the commonly used airfoil for the fixed landing gear would be E864 airfoil. One of the commonly available aluminium alloy for fabrication of landing gears is Al6061-T6. Yield stress of this variety of aluminium comes at around 276 MPa. Assuming a factor of safety of 2.5, we get

$$\sigma_{yield} = \frac{276}{2.5} = 110.4MPa$$

where, σ_{yield} is the yield stress of material.
Hence Maximum stress experienced by nose landing gear and main landing gear should not exceed this value.

$$Area_{\mathrm{Nose_{min}}} = \frac{Load_{\mathrm{Nose}}}{\text{Yield Stress}}$$
$$Area_{\mathrm{Nose_{min}}} = \frac{10.19}{110.4 \times 10^6} = 0.0923\,\mathrm{mm}^2$$

Similarly for main landing gear, we do the same landing calculations,

$$Area_{main} = 0.196\ mm^2 \tag{8.1}$$

Using the Eppler airfoil parameters, we can choose the landing strut Chord of the airfoil to be

$$C_{\mathrm{Nose_{min}}} = 1\ mm \tag{8.2}$$
$$C_{\mathrm{Main_{min}}} = 2\ mm \tag{8.3}$$

where, $C_{\mathrm{Nose_{min}}}$ and $C_{\mathrm{Main_{min}}}$ are the minimum chord lengths of nose and main landing gear struts.
Since the minimum values chord values are 1mm and two mm respectively, we can safely take anywhere from 5 to 10mm as our chord length for our UAV. we will be going with Chord of 10mm which gets the design that has light drag.

8.9 Tire sizing

The tires are sized to carry the weight of the aircraft. For a small fixed wing UAV, a solid, non-inflatable tire made of foam or plastic is generally used.

8.9.1 Diameter and Width of Tire

As provided in [10], the diameter and width of main tires can estimated using,

$$d = A{W_w}^b$$

where d is diameter(cm), W_w is weight on the wheel, $A = 5.1$ and $b = 0.349$ for a general aviation aircraft. Since, for a smaller fixed wing UAV, we need considerably lower diameter of tire, we take $A = 2.75$ **and** $b = 0.25$. we can get W_w by,

$$2W_w = \frac{B_n}{B}W$$

Weight of our aircraft is 6.51 kg from second weight estimate, B = 0.55 m and $B_n = 0.1045$ m.

$$W_w = 21.68N$$

This is also the **max static load of the main wheel.**

$$d = 2.75 \times (21.68)^{0.25}$$

$$d \approx 6cm.$$

Examining tire data from [10], we get that the ratio of width to diameter of tires is roughly 0.33. So using this ratio as a preliminary estimate, we get width, w of tire = $0.33 \times d = 1.98cm$.

Therefore the diameter and width of main tire is 6 cm and 1.98 cm respectively.

The nose tires can be about 60-100 % of the main tire size. So the diameter and width of nose landing gear can be from 3.6 - 6 cm and 1.19 - 1.98 cm respectively.

8.9.2 Static and Dynamic Loading

The static loads on each main wheel and nose gear wheel was already found to be 27.145 N and 9.58 N respectively.

[11] gives the following forumlas for calculation of dynamic loads.

For nose landing gear,

$$F_{n_{dyn}} = \frac{a_L W H_{cg}}{gB}$$

for main landing gear,

$$F_{m_{dyn}} = \frac{a_T W H_{cg}}{gB}$$

where, a_T and a_L are the magnitudes of takeoff and landing accelerations respectively, H_{cg} is the height of the cg from the ground and W is the weight of the aircraft.

For our UAV, we know W = 5.47g.
The cg can be assumed to be at the centre of the overall height of the aircraft for the time being.So,

$$H_{cg} = H + 0.5 \times H_{fuselage} = 21.56 + 0.5 \times 13.66 = 28.39cm$$

We can assume a typical braking coefficient of 0.3, which gives a deceleration of 0.3g. So, we can assume an acceleration while takeoff as $4m/s^2$. These are an upper limit. So, we get an upper bound of dynamic load. So,

$$F_{n_{dyn}} = 0.3 \times 5.47 \times 9.81 \times \frac{28.39}{55} = 8.31N$$
$$F_{m_{dyn}} = 4 \times 5.47 \times \frac{28.39}{55} = 11.29N$$

Net load is a summation of static and dynamic loads.

$$F_n = 10.19 + 8.31 = 18.5N$$
$$F_m = 43.46 + 11.29 = 54.75N$$

8.10 Compilation of Parameters

Parameter	**Value**
Wheel Base	55 cm
Wheel Track	20.1 cm
Landing Gear Height	21.56 cm
Clearance Angle	16°
Tire Diameter	6 cm
Tire Width	1.98 cm

8.11 Shork Absorbers and Steerability

The nose-wheel will have to be made steerable to account for gusts on the run-way. This would ensure the possibility for course correction in the event that control needs to be regained during ground roll.
Shock absorbers are required to damp out the landing gear so that ground roll will not have negative impacts on the UAV body. A simple spring system is sufficient given that the UAV is comparatively light in weight.
Steerability and damping mechanisms will be taken care of during fabrication.

Chapter 9

CG Location and Internal Layout

9.1 Weights of Components

We calculate the theoretical weight of each component and also obtain the weight of the component from the CAD model in fusion 360. We used a thickness of 0.3 mm for the material and will multiply a factor of 1.5 to the weight obtained from CAD to account for the weight of the structural components like ribs, spars etc.

9.1.1 Wing

Theoretical weight

From [11], we have,

$$W_w = S_w \cdot \text{MAC} \cdot \left(\frac{t}{c}\right)_{\text{max}} \cdot \rho_{\text{mat}} \cdot K_\rho \cdot \left(\frac{AR \cdot n_{\text{ult}}}{\cos(\Lambda_{0.25})}\right)^{0.6} \cdot \lambda^{0.04} \cdot g$$

where,

1. $(\frac{t}{c})_{max}$ the maximum thickness-to-chord ratio for airfoil = 0.12 for NACA2412.
2. K_ρ is the wing density factor = 0.0012 ([11]).
3. $n_{ult} = 1.5 n_{max}$ is the ultimate load factor. We take $n_{max} = 2$ and so $n_{ult} = 3$.
4. We use Aluminium for fabrication. So, $\rho_{mat} = 2700 Kg/m^3$.

Other parameters are known from wing design.
Substituting values, we get $W_w = 1.099 Kg$

From CAD model

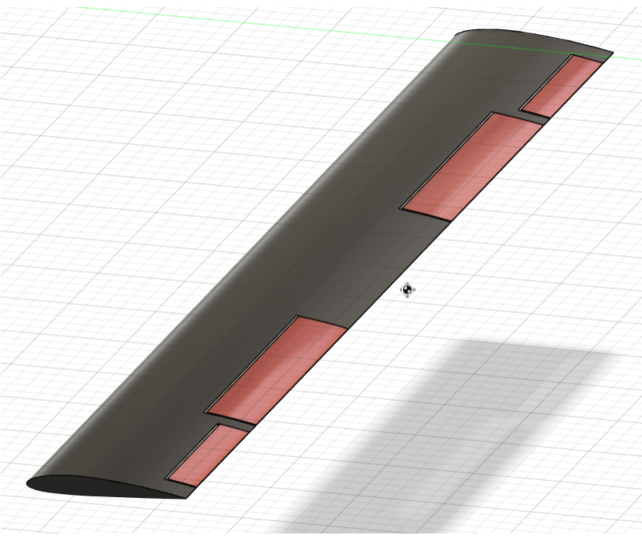

Figure 9.1: CAD model of wing

The overall weight of the wing including ailerons and flaps comes out to be 696.232 g = 0.635 Kg. So, the weight is taken as $1.5 \times 0.952 = 0.952$ Kg.

9.1.2 Horizontal Tail

Theoretical

[11] gives the formula for estimating the Horizontal tail weight.

$$W_{HT} = S_{HT} \cdot \mathrm{MAC}_{HT} \cdot \left(\frac{t}{c}\right)_{\mathrm{max}_{HT}} \cdot \rho_{\mathrm{mat}} \cdot K_{\rho_{HT}} \cdot \left(\frac{AR_{HT}}{\cos(\Lambda_{0.25_{HT}})}\right)^{0.6} \cdot \lambda_{HT}^{0.04} \cdot V_H^{0.3} \cdot \left(\frac{C_e}{C_T}\right)^{0.4}$$

where, $(\frac{t}{c})_{max}$ the maximum thickness-to-chord ratio for airfoil = 0.12 for NACA0012.

- $K_{\rho_{HT}}$ is the horizontal tail density factor = 0.02 ([11]).
- $\frac{C_e}{C_T}$ is the ratio of chord of elevator to the chord of horizontal tail. We take a standard value of 0.3.

Remaining values are known from Tail design.
Substituting values, we get $W_{HT} = 0.5856Kg$.

From CAD Model

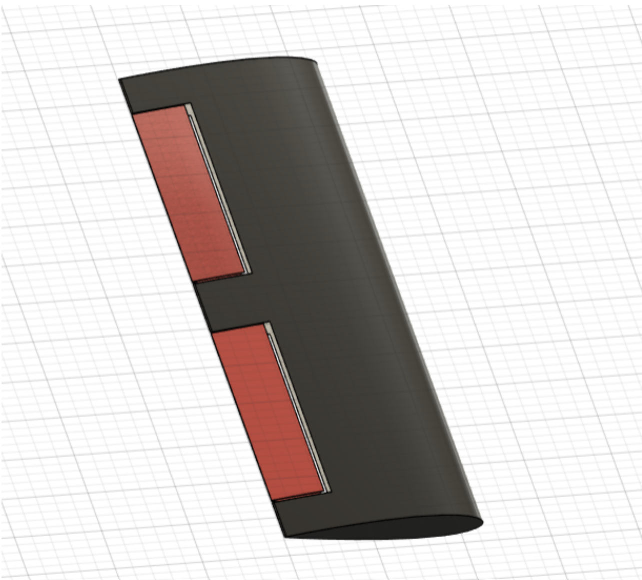

Figure 9.2: CAD model of horizontal tail

The overall weight of the Horizontal Tail including elevators comes out to be 110.406 g = 0.11 Kg. So, the weight is considered to be $1.5 \times 0.11 = 0.165$ Kg.

9.1.3 Vertical Tail

Theoretical

[11] gives the formula for estimating the Vertical tail weight.

$$W_{VT} = S_{VT}.MAC_{VT}.\left(\frac{t}{c}\right)_{max_{VT}}.\rho_{mat}.K_{\rho VT}.\left(\frac{AR_{VT}}{cos(\Lambda_{0.25_{VT}})}\right)^{0.6}.\ \lambda_{VT}^{0.04}.\ \overline{V}_{VT}^{0.2}\ (\frac{C_r}{C_V})^{0.4}$$

where, $(\frac{t}{c})_{max}$ the maximum thickness-to-chord ratio for airfoil = 0.12 for NACA0012.

- $K_{\rho_{VT}}$ is the vertical tail density factor = 0.06 ([11]).
- $\frac{C_r}{C_V}$ is the ratio of chord of rudder to the chord of vertical tail. We take a standard value of 0.4.

Remaining values are known from Tail design.
Substituting values, we get $W_{HT} = 0.1629Kg$..

From CAD model

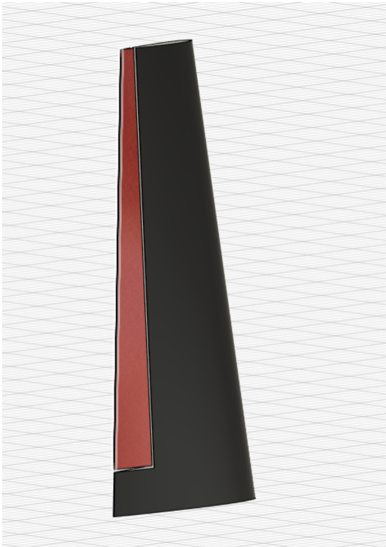

Figure 9.3: CAD model of vertical tail

The overall weight of the Vertical Tail including rudder comes out to be 64.479 g = 0.064 Kg. So, the weight is taken as $1.5 \times 0.064 = 0.096$ Kg.

9.1.4 Fuselage

Theoretical

[11] gives the formula for estimating weight of fuselage.

$$W_F = L_f \cdot D_{f_{\max}}^2 \cdot \rho_{\text{mat}} \cdot K_{\rho_f} \cdot n_{\text{ult}}^{0.25} \cdot K_{\text{inlet}}$$

where,

- L_f is the fuselage length = 1.69 m and D_f is the maximum diameter of fuselage. For a rectangular fuselage this would be the width. So, $D_f = 0.1681$ m.
- K_{ρ_f} represents the fuselage density factor = 0.003 ([11]).
- K_{inlet} is 1.25 for the case of inlets on the fuselage ([11]) and $n_{ult} = 3$ as before.

Substituting all known values, we get $W_F = 0.64$ Kg.

From CAD model

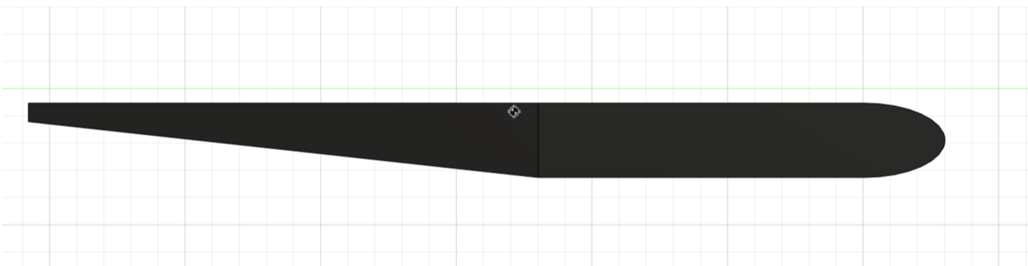

Figure 9.4: side view of CAD model of fuselage

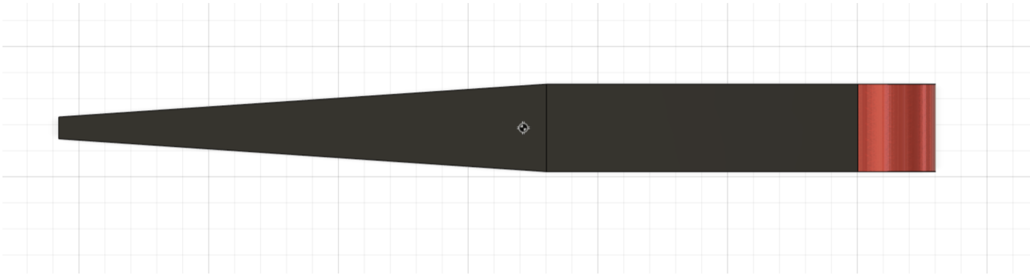

Figure 9.5: Top view of CAD model of fuselage

The overall weight of the fuselage including rudder comes out to be 658.326 g = 0.658 Kg. So, the weight is taken as $1.5 \times 0.658 = 0.987$ Kg.

9.1.5 Landing Gear

Theoretical

[11] gives the formula for estimating the Landing Gear weight.

$$W_{LG} = K_L \cdot K_{ret} \cdot K_{LG} \cdot W_L \cdot \left(\frac{H_{LG}}{b}\right) \cdot \eta_{ult_{land}}^{0.2}$$

- K_L is the landing place vector = 1 ([11]).
- K_{ret} is 1 for fixed landing gear.
- K_{LG} is landing gear weight factor = 0.52

Remaining values are known from Second weight estimation, Landing gear design and Wing Selection. Substituting values, we get $W_{LG} = 0.4584Kg$.

From CAD model

Figure 9.6: CAD model of landing gear

The weight of the landing gear comes out to be 492.56 g.

9.1.6 Electronics

Battery, camera, motors and propellers are already mentioned in previous chapters. The other components required are given here.

1. **Electronic Speed Controller (ESC)** - 20g

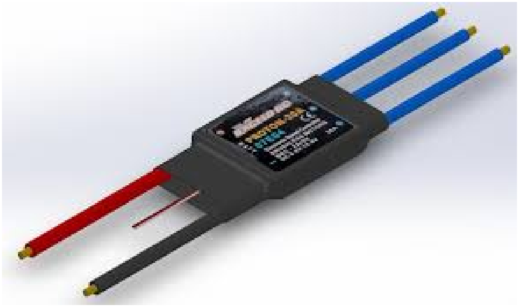

Figure 9.7: ESC

2. **Servos -** 9 g each. We use 5 of these, 1 flaps, 2 ailerons 1 rudder and 1 elevator.

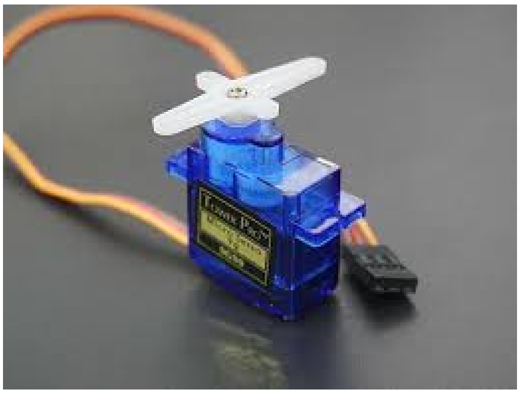

Figure 9.8: Servo

3. **Pixhawk** - 60 g

Figure 9.9: Pixhawk

4. **Reciever** - 15 g

Figure 9.10: Reciever

9.2 CAD Model of the UAV

Reciever and ESCs are not shown in the CAD. They will be placed close to the CG and will not have a significant effect on the CG location.

9.2.1 CAD Model not showing internal components

The CG point is marked in the CAD model

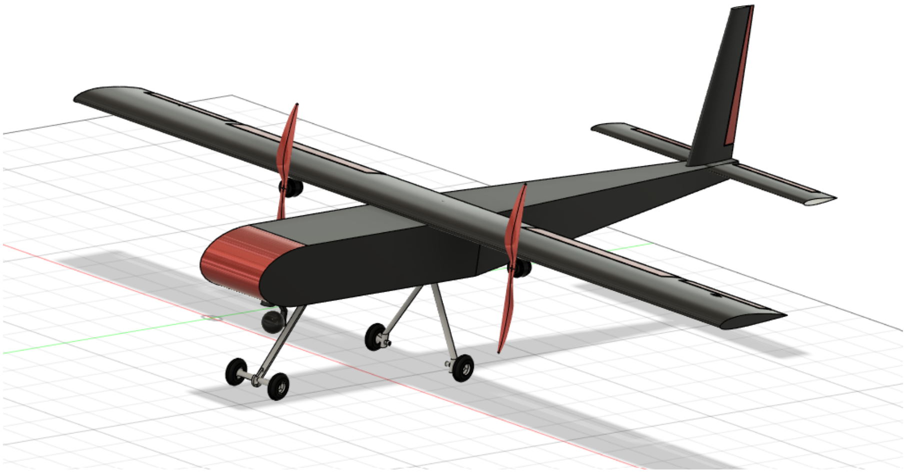

Figure 9.11: CAD model of UAV

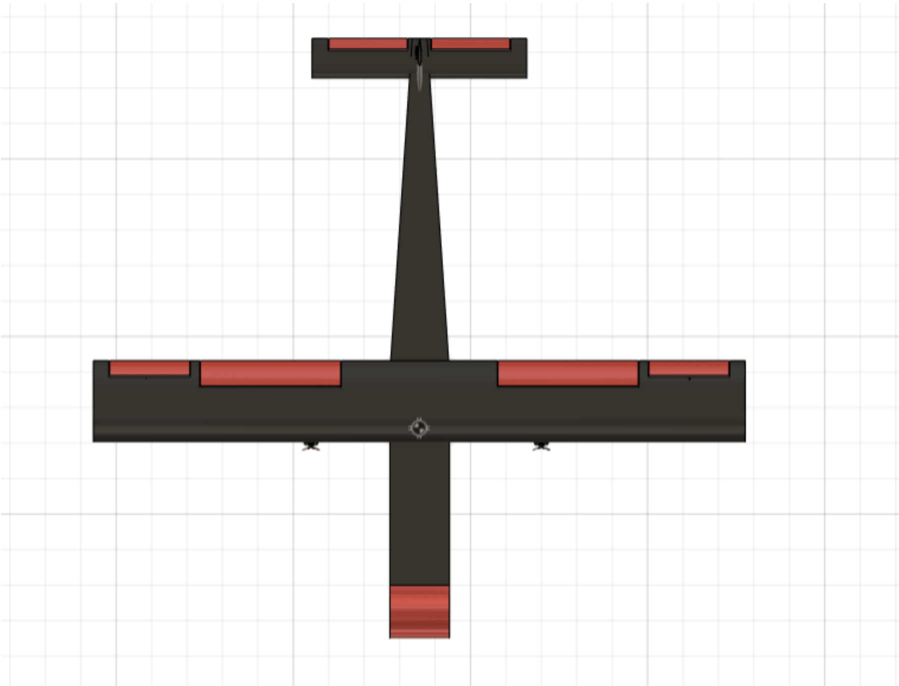

Figure 9.12: Top view

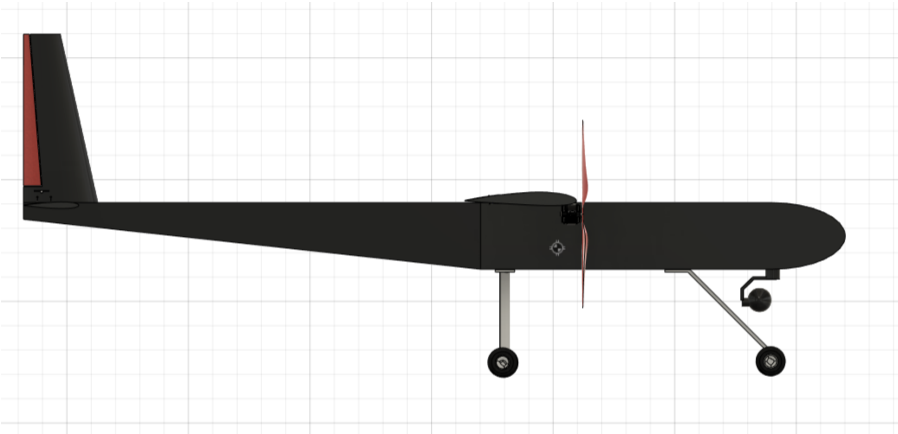

Figure 9.13: Side view

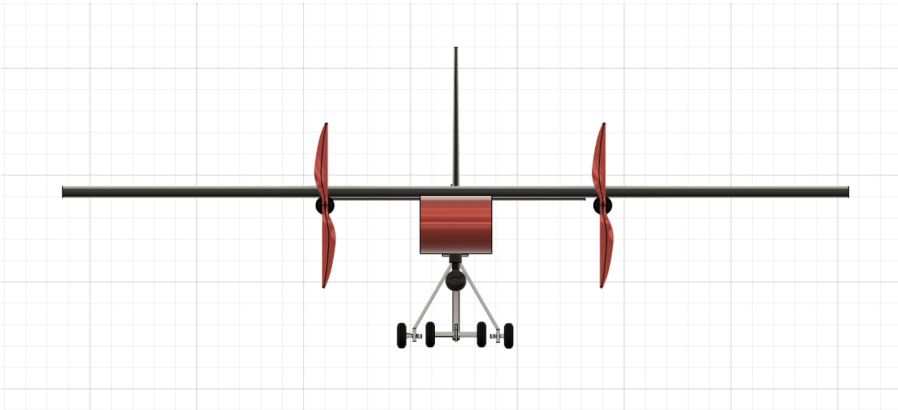

Figure 9.14: Front view

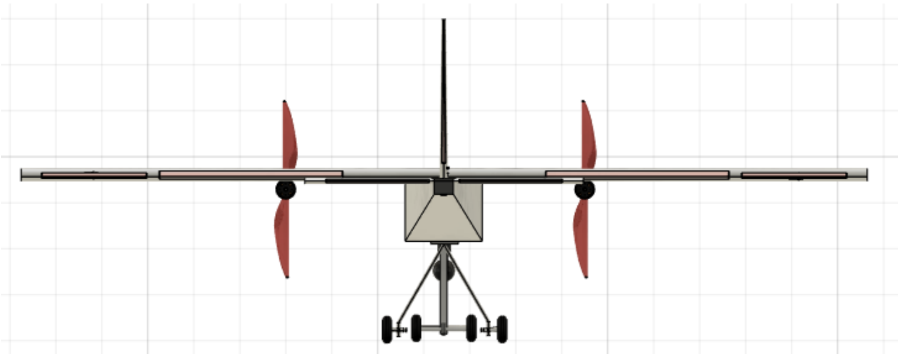

Figure 9.15: Back view

9.2.2 CAD model showing internal layout

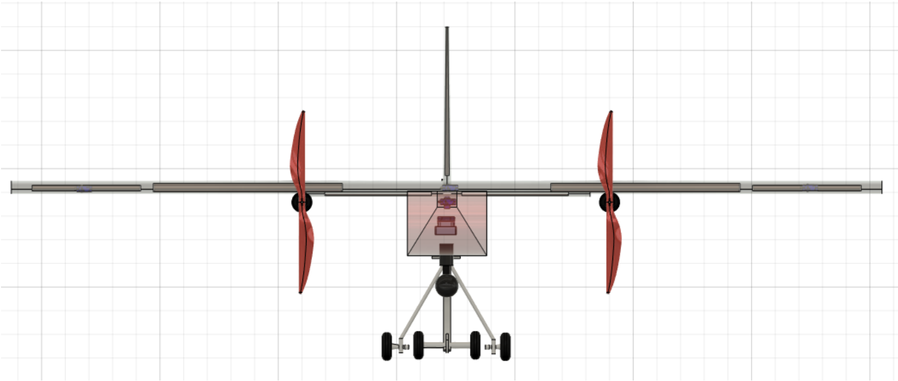

Figure 9.16: Front view showing camera and propellers

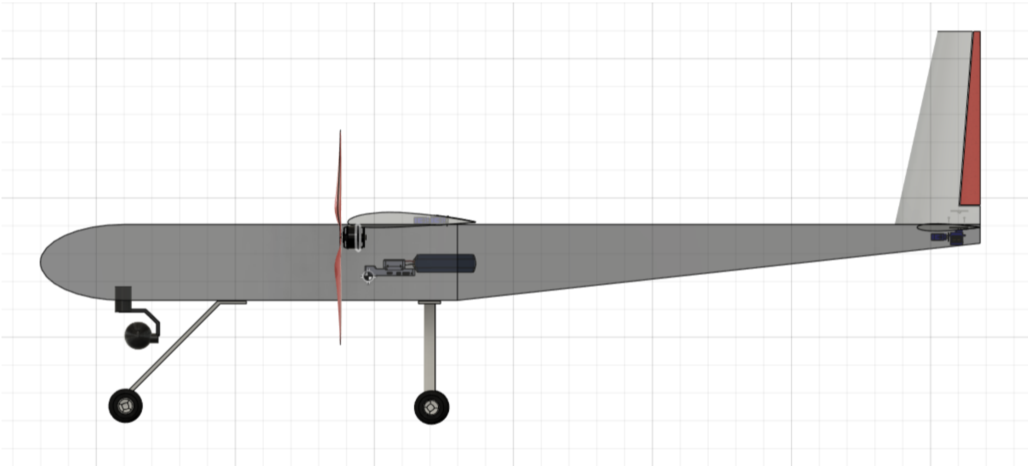

Figure 9.17: Side view showing components

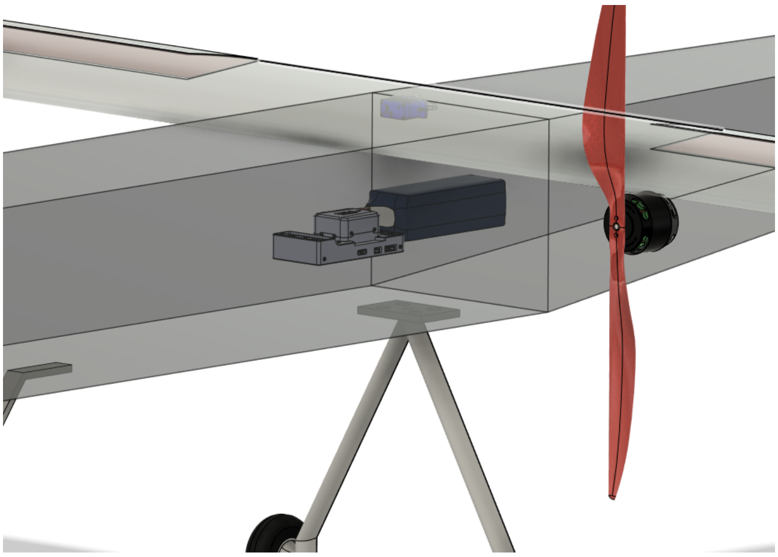

Figure 9.18: Battery and pixhawk placement

9.3 Weight Estimation from CAD

Component	Number	Weight
Battery	1	1.25 Kg
Camera	1	0.8 Kg
Servo	5	45 g
Reciever	1	15 g
Motor	2	390 g
Propeller	2	44 g
Pixhawk	1	60 g
ESC	2	40 g

Net weight of this set is 2.644 Kg.

Component	Weight
Fuselage	987.49 g
Wing	952.22 g
Horizontal Tail	165.61 g
Vertical Tail	96.72 g
Landing Gear	492.56 g

Net weight of this set is 2.695 Kg.
The total weight of UAV comes out to be 5.288 Kg.

9.4 Location Of CG

We can find the location of CG by taking a weighted average of all components as we know where they are placed. The location of CG is found directly from the CAD model in Fusion360.

x_{cg} is the distance of CG from the nose of the UAV in in the longitudinal direction. y_{cg} is the distance of CG from the base of the fuselage in the direction perpendicular to the ground. z_{cg} is the location of CG from the side wall of the fuselage in the direction perpendicular to it.

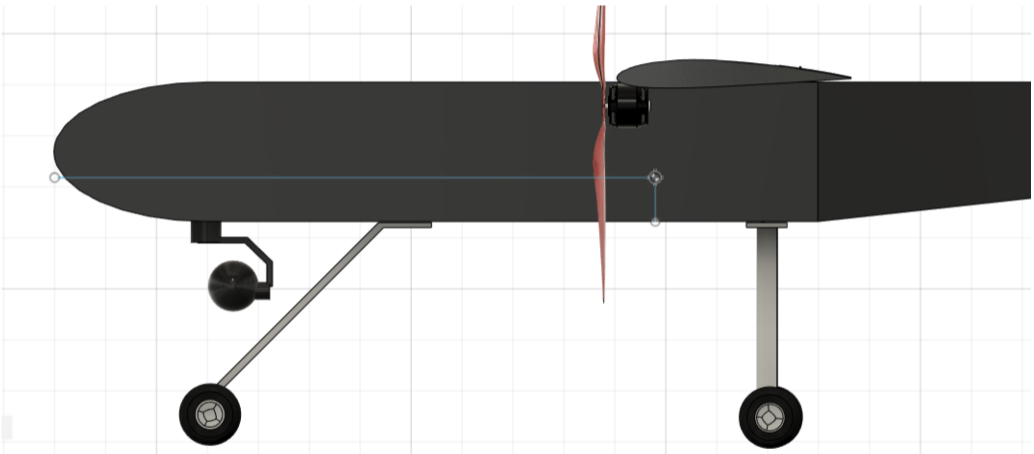

Figure 9.19: Location of x_{cg} and y_{cg}

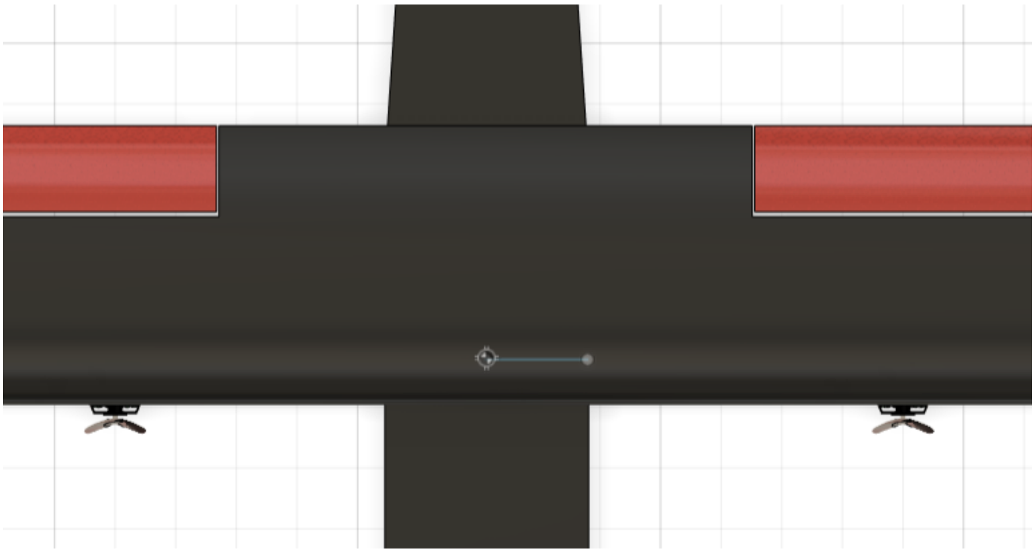

Figure 9.20: Location of z_{cg}

$x_{cg} = 63$cm, $y_c g = 5.26$ cm, $z_{cg} = 8.405$ cm.

Chapter 10

Stability Analysis

10.1 Longitudinal Stability

10.1.1 Pitching Moment Equation

The major contributors of aircraft pitching moment about c.g are wing, fuselage, tail and engine.

A wing's lift through its aerodynamic center and its moment about that center make up the wing pitching-moment contribution. the aerodynamic center is typically at 25% of the MAC in subsonic flight i.e quarter chord. Drag of the wing and tail produces some pitching moment, but these values are negligibly small. Also, the pitching moment of the tail about its aerodynamic center is small and can be ignored.

To trim and control the aircraft, a very big moment is produced by multiplying the long moment arm of the tail by its lift. The fuselage and nacelles produce pitching moments and are influenced by the upwash and downwash produced by the wing. The engine also produces various pitching moments but the major contributor is the Thrust times its vertical distance from c.g.

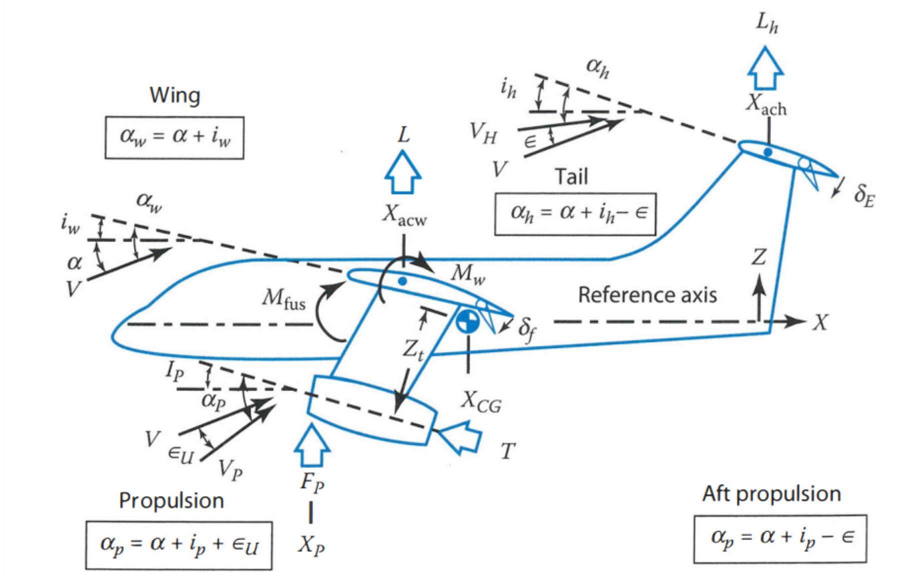

Figure 10.1: Longitudinal moments, taken from [10]

The sum of Moments about c.g is given by,

$$M_{cg} = L(X_{cg} - X_{ac}) + M_w + M_{w\delta f}\delta f + M_{fus} - L_h(X_{ach} - X_{cg}) - TZ_t + F_p(X_{cg} - X_p) \tag{10.1}$$

where, M_{cg} is total moment about cg, X_y is distance of point y from nose, M_w is moment of wing about cg, M_{fus} is moment of fuselage about cg, $TZ_t$is thrust produced by propellers times it vertical distance from cg and F_p is vertical force produced at propeller disk which is negligible and is equated to 0.

we can get the coefficient form by dividing the above equation by qS_wc,

$$Cm_{cg} = C_L(\frac{X_{cg} - X_{ac}}{c}) + C_{m_w} + C_{m_{w\delta f}}\delta f + C_{m_{fus}} - \eta_h\frac{S_h}{S_w}C_{L_h}(\frac{X_{ach} - X_{cg}}{c}) - \frac{TZ_t}{qS_wc} \tag{10.2}$$

where $\eta_h = \frac{q_h}{q}$ is ratio between the dynamic pressure at the tail and the freestream dynamic pressure which is 0.9 for our uav.

10.1.2 Static Pitch Stability

Any change in angle of attack must produce moments that oppose the change is needed for static pitch stability. In other words, the derivative of pitching moment with respect to angle of attack must be negative i.e

$$\frac{\partial C_m}{\partial \alpha} < 0$$

Because of downwash effects, the tail angle of attack does not vary directly with aircraft angle of attack. So, a derivative term is added to account for this. We ignore the propeller normal force, F_p and write:

$$Cm_\alpha = C_{L_\alpha}(\bar{X}_{cg} - \bar{X}_{ac}) + C_{m_{\alpha fus}} - \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha}(\bar{X}_{ach} - \bar{X}_{cg}) \tag{10.3}$$

where,
$\bar{X}_y$ is X_y/c.
$\frac{\partial \alpha_h}{\partial \alpha}$ = 1 - $\frac{\partial \epsilon}{\partial \alpha}$, $\epsilon = \epsilon_0 + \epsilon_\alpha \alpha$.
where,

$$\epsilon_0 = \frac{2C_{L_{cruise}}}{\pi AR_w} = 0.042$$

$$\epsilon_\alpha = \frac{2C_{L_{\alpha_w}}}{\pi AR_w} = 0.542 \; per \; radian$$

$$\implies \frac{\partial \alpha_h}{\partial \alpha} = 1 - 0.542 = 0.458$$

[10] mentions formula for finding $C_{m_{fus}}$:

$$C_{m_{fus}} = \frac{K_f W_{fus}^2 L_f}{\bar{c} S_w} \; per \; degree$$

where, where K_{fus} is empirical pitching moment of fuselage and is 0.01 for our uav from the figure shown below, W_f is width of fuselage and L_f is length of fuselage.

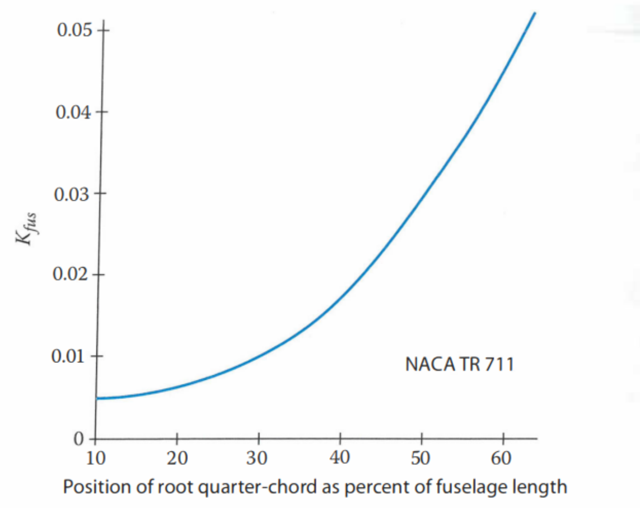

Figure 10.2: Fuselage moment term, taken from [10]

The x-axis term comes out to be, $\frac{61.75}{169} \times 100\% = 33.65\%$. Corresponding to it, we get K_{fus} of about 0.015. With these, we get $C_{m_{\alpha_{fus}}} = 0.004943$ per degree = 0.283 per radian.

Substituting values,

$$C_{m_\alpha} = 6.81 \times \frac{(63 - 65.04)}{23} + 0.425 - 0.9 \times \frac{0.069}{0.42} \times 6.79 \times 0.458 \times \frac{157.59 - 63}{23} = -2.069 \; per \; radian$$

$C_{m_\alpha} < 0$ as required for longitudinal stability.

The pitching-moment derivative's magnitude varies with the location of the c.g. For any aircraft there is a c.g. location that provides no change in pitching moment as angle of attack is varied. The most-aft c.g. position before the aircraft becomes unstable is this aerodynamic center of the aircraft, also known as the neutral point (X_{np}), which denotes neutral stability.

We can solve for $\bar{X}_{np}$ in above equation by putting $Cm_\alpha = 0$.

$$\bar{X}_{np} = \frac{C_{L_\alpha}\bar{X}_{ac} - C_{m_{\alpha fus}} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} \bar{X}_{ach}}{C_{L_\alpha} + \eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha}} \tag{10.4}$$

- $C_{L_\alpha}\bar{X}_{ac} = 6.8067 \times \frac{65.04}{23} = 19.248$
- $C_{m_{\alpha fus}} = \frac{k_{fus} W_f^2 L_f}{c S_w}$ per degree.
  where K_{fus} is empirical pitching moment of fuselage and is 0.015 for our uav as per [10], W_f is width of fuselage and L_f is length of fuselage.
  $$C_{m_{\alpha fus}} = \frac{0.015 \times 0.1681^2 \times 1.69}{0.23 \times 0.42} = 0.0074 \; per\ degree = 0.425 \; per\ radian$$
- $\eta_h \frac{S_h}{S_w} C_{L_{\alpha_h}} \frac{\partial \alpha_h}{\partial \alpha} = 0.9 \times \frac{0.069}{0.42} \times 6.79 \times 0.458 = 0.46$
- $\bar{X}_{ach} = 157.59/23 = 6.85$

Now,

$$\bar{X}_{np} = \frac{19.248 - 0.425 + 0.46 \times 6.85}{6.81 + 0.46} = 3.022$$
$$X_{np} = 3.022 \times c = 69.52 \; cm$$

Therefore neutral point is at a distance of 69.52 cm from the nose of the aircraft.

10.1.3 Static Margin

The distance in percent MAC from the neutral point to the c.g is called Static margin. The most crucial concept for an aircraft's longitudinal stability is the static margin. It can also be computed as the lift coefficient derivative divided by the pitching-moment derivative.

$$Static\ Margin(SM) = \bar{X}_{np} - \bar{X}_{cg} \tag{10.5}$$

The aircraft is stable if c.g is ahead of the neutral point (positive static margin) as it gives negative pitching moment derivative as per the equation above.

The static margin of our aircraft is,

$$SM = 3.022 - 2.739 = 0.283$$

10.1.4 Trim Condition

The total pitching moment needs to equal zero in order to have a static trim condition.i.e $C_{m_{cg}} = 0$. We ignore the contribution of fuselage and setting flap deflection to zero for static trim,

$$Cm_{cg} = C_L\left(\frac{X_{cg} - X_{ac}}{c}\right) + C_{m_w} - \eta_h \frac{S_h}{S_w} C_{L_h}\left(\frac{X_{ach} - X_{cg}}{c}\right) - \frac{TZ_t}{qS_w c}$$

- $C_L = C_{L_0} + C_{L_\alpha}\alpha_{trim}$.
  Substituting known values, we get $\alpha_{trim} = 2.502°$
- $C_{L_h} = C_{L_{\alpha h}}(\alpha + i_h - \epsilon - \alpha_{0L_h}) = 6.79(\alpha + 0 - (0.042 + 0.542\alpha) - 0) = -0.285 + 3.11\alpha$
- From [10],
  $$C_{m_w} = C_{m_{0_{\text{airfoil}}}}\left(\frac{A\cos^2\Lambda}{A + 2\cos\Lambda}\right)$$
  for an unswept wing, with $C_{m_{0_{airfoil}}}$ = -0.05, C_{m_w} = -0.04.
- $\frac{TZ_t}{qS_w c} = \frac{4.72 \times 0.082}{0.5 \times 1.2 \times 20^2 \times 0.42 \times 0.23} = 0.017$

Using these values and substituting in the trim equation, we get $C_{L_h} = 0.3413$. Using this and other known values in $C_{L_h} = C_{L_{\alpha h}}(\alpha + i_h - \epsilon - \alpha_{0L_h})$ and solving for i_h, we get the tail incidence angle, $i_h = -0.012°$.

10.2 Lateral Stability

The lateral-directional analysis really embraces two closely coupled analyses: the yaw (directional) and the roll (lateral). Both directional and lateral analysis depend on the yaw angle, β and do not really depend on the roll angle, ϕ. Hence, the term "lateral" is used to refer to both lateral and directional stability analyses.

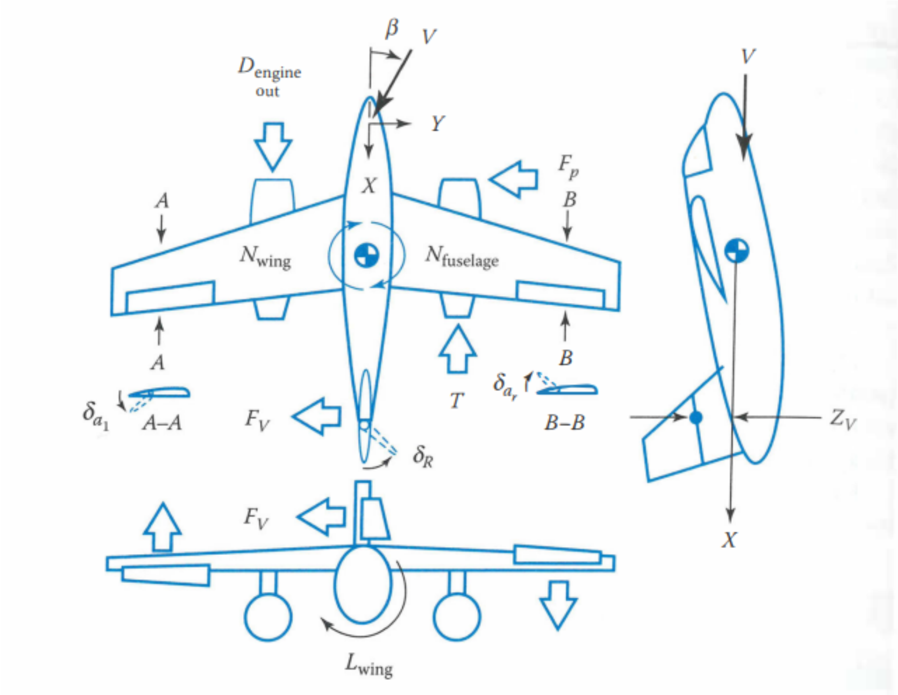

Figure 10.3: Lateral Geometry, taken from [10]

We use N to represent the Yawing moment and L to represent Rolling moment. The sign convention is taken such a way that both L and N are positive to the right.

$$C_n = \frac{N}{qS_w b}$$
$$C_l = \frac{L}{qS_w b}$$

where q is the dynamic pressure, S_w is the wing area, b is the wing span.
We require:

$$C_{n_\beta} > 0$$
$$C_{l_\beta} < 0$$

10.2.1 Yawing Moment

The major yawing moment is due to the lateral lift of the vertical tail, denoted by F_v. This counteracts the fuselage yawing moment, which is generally negative to the sense shown in the figure above. Rudder deflection acts as a flap to increase the lateral lift of the vertical tail. Another wing yawing moment occurs with aileron deflection. The wing with increased lift due to aileron deflection has more induced drag, so the yawing moment is in the opposite direction from the rolling moment due to the aileron deflection. This is known as "adverse yaw."

The derivative of C_n with the side-slip angle, β is given by,

$$C_{n_\beta} = C_{n_{\beta_w}} + C_{n_{\beta_{fus}}} + C_{n_{\beta_v}} - \frac{F_{p\beta}}{qS_w}\frac{\partial \beta_p}{\partial \beta}(\bar{X}_{cg} - \bar{X}_p)$$

where, $C_{n_{\beta_w}}$ is due to the wing, $C_{n_{\beta_{fus}}}$ is due to the fuselage and $C_{n_{\beta_v}}$ is due to the fuselage. The fourth term is due to propwash and is difficult to predict. Since, we don't have very high rpm of props, we are neglecting this term.
Another term we would require is the C_{F_β} which is actually similar to C_{L_α} for a lateral force and magnitude

will be the same. The lengths which are shown with overhead bars are actually expressed as fractions of the wing span b.

Now, we will compute the contribution of each of these terms towards the net C_{n_β} by following the formulas given in [10].

1. **Due to Vertical Tail**

$$C_{n_{\beta_v}} = C_{F_{\beta_v}} \frac{\partial \beta_v}{\partial \beta} \eta_v \frac{S_v}{S_w} (\bar{X}_{ac_v} - \bar{X}_{cg})$$

We have NACA0012 as the airfoil for our vertical tail. $C_{L_\alpha} = 0.1197$ per degree. The $C_{L_{\alpha_v}}$ for the vertical tail is then given by,

$$C_{L_{\alpha_v}} = \frac{C_{L_\alpha}}{1 + \frac{C_{L_\alpha}}{\pi e AR}}$$

So, we get $C_{F_{\beta_v}} = 0.1186$ per degree or 6.795 per radian.
$\frac{\partial \beta_v}{\partial \beta}$ is the side-slip derivative and η_v is the local dynamic pressure ratio and side-slip derivative and have the relation,

$$\frac{\partial \beta_v}{\partial \beta} \eta_v = 0.724 + \frac{3.06 \frac{S'_{vs}}{S_w}}{1 + cos\Lambda} - 0.4 \frac{Z_{wf}}{D_f} + 0.009 A_{wing}$$

S'_{vs} is the area of the vertical tail $= 0.04\ m^2$, Λ is the dihedral angle $= 0°$, Z_{wf} is the vertical height of wing above fuselage center-line $= \frac{13.66}{2}$cm, D_f is the depth of the fuselage $= 13.66$ cm, A_{wing} is the Aspect Ratio of wing $= 8$.
We assuming $\eta_v = 0.9$ and substituting the values, we get $\frac{\partial \beta_v}{\partial \beta} = 0.824$.
The distance between the CG and the AC of vertical tail is found out from CAD model and comes out to be 98.4 cm.
Substituting all of these into the formula we get,

$$C_{n_{\beta_v}} = 6.795 \times 0.824 \times 0.9 \times \frac{0.04}{0.42} \times \frac{98.4}{184} = 0.2566\ per\ radian.$$

2. **Due to Fuselage**

$$C_{n_{\beta_{fus}}} = -1.3 \frac{V_f}{S_w b} \frac{D_f}{W_f}$$

V_f is the volume of fuselage $= 25915.87\ cm^3$, W_f is the width of the fuselage $= 16.81$ cm. Substituting these values with other known values, we get,

$$C_{n_{\beta_{fus}}} = -1.3 \times \frac{0.0259}{0.42 \times 1.84} \frac{13.66}{16.81} = -0.0354\ per\ radian.$$

3. **For wing**

$$C_{n_{\beta_w}} = C_L^2 \left(\frac{1}{4\pi A} - \left[\frac{\tan\Lambda}{\pi A(A + 4\cos\Lambda)} \right] \times \left[\cos\Lambda - \frac{A}{2} - \frac{A^2}{8\cos\Lambda} + \frac{6(\overline{X}_{ac_w} - \overline{X}_{cg})\sin\Lambda}{A} \right] \right)$$

For dihedral of 0°, Aspect Ratio of wing, A = 8 and $C_L = 0.532$, we get,

$$C_{n_{\beta_w}} = \frac{C_L^2}{4\pi A} = 0.0028\ per\ radian.$$

Adding all three, we get,

$$C_{n_\beta} = 0.2566 - 0.0354 + 0.0028 = 0.224\ per\ radian.$$

We find that $C_{n_\beta} > 0$ as required for directional stability.

10.2.2 Rolling Moment

The major components affecting the rolling moment of an aircraft are the wing and the vertical tail.

1. **Due to Vertical Tail**

$$C_{l_{\beta_v}} = -C_{F_{\beta_v}} \frac{\partial \beta_v}{\partial \beta} \eta_v \frac{S_v}{S_w} \bar{Z}_v$$

Z_v is the vertical distance between the CG of the aircraft and the CG of the vertical tail. From the CAD model, we find $Z_v = 24.68$ cm.
Substituting the other known values, we get,

$$C_{l_{\beta_v}} = -6.795 \times 0.824 \times 0.9 \times \frac{0.04}{0.42} \times \frac{24.68}{184} = -0.0644\ per\ radian.$$

2. **Due to Wing** We can consider the effect of wing on the rolling moment as two parts - due to wing vertical placement on the fuselage ($C_{l_{\beta_{wf}}}$) and due to due to the sweep ($C_{l_{\beta_\Lambda}}$), which would be 0 as we don't have sweep for our wing. We add these two values to the contribution of wing dihedral effect ($C_{l_{\beta_{wing}}}$) to obtain the net $C_{l_{\beta_w}}$.

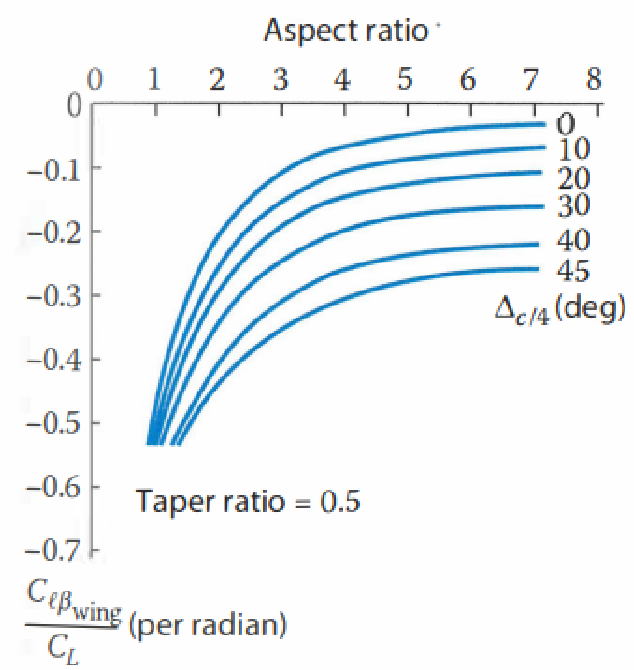

Figure 10.4: Dihedral effect of aspect ratio, taper ratio, and sweep, taken from [10]

From the above figure, we see that for a wing of Aspect ratio = 8, unswept and taper ratio of 0.5, the contribution of dihedral towards the net $C_{l_{\beta_w}}$ is negligible. Hence, $C_{l_{beta_{wing}}} = 0$.
[10] gives the formulas for finding the other components of $C_{l_{\beta_w}}$. Due to sweep,

$$C_{l_{\beta_\Gamma}} = -\frac{C_{L_\alpha}\Gamma}{4}\left[\frac{2(1+2\lambda)}{3(1+\lambda)}\right] = 0 \quad as \;\; \Gamma = 0$$

Due to wing placement on fuselage,

$$\begin{aligned} C_{l_{\beta_{wf}}} &= -1.2\frac{\sqrt{A}Z_{wf}(D_f+W_f)}{b^2} \\ &= -1.2\times\sqrt{8}\times\frac{0.1366}{2}\times\frac{(0.1366+0.1681)}{(1.84)^2} \\ C_{l_{\beta_{wf}}} &= -0.021 \; per \; radian. \end{aligned}$$

Summing up,

$$C_{l_{\beta_w}} = (\frac{C_{l_{\beta_{wing}}}}{C_L})C_L + C_{L_{\beta_\Gamma}} + C_{l_{\beta_{wf}}} = -0.021 \; per \; radian.$$

Now writing C_{l_β} as the sum of both components,

$$C_{l_\beta} = C_{l_{\beta_w}} + C_{l_{\beta_v}} = -0.0644 - 0.021 = -0.0854 \; per \; radian.$$

$C_{l_\beta} < 0$ as required for lateral stability.

10.3 Control Surface sizing

10.3.1 Aileron Design

The primary function of an aileron is the lateral (i.e., roll) control of an aircraft; however, it also affects the directional control. For this reason, the aileron and the rudder are usually designed concurrently. Lateral control is governed primarily through a roll rate (P).

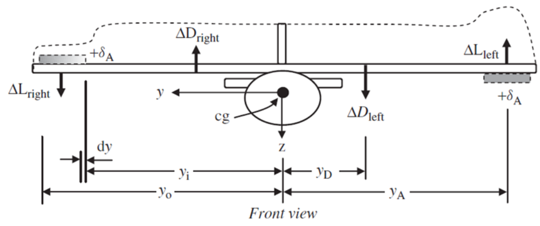

Figure 10.5: Rear view of aircraft

The typical values for parameters for aileron design are as follows:
$\frac{S_a}{S} = 0.05 - 0.1$, $\frac{b_a}{b} = 0.2 - 0.3$, $\frac{c_a}{c} = 0.15 - 0.25$ and span ratio of 0.6 - 0.8.
Due to lack of further information at this point, we choose the average value of chord and span to find the aileron geometry.

$$\begin{aligned} b_a &= 0.25b = 0.46m \\ c_a &= 0.2c = 0.046m \\ b_{a_i} &= 0.7b = 1.288m \end{aligned}$$

For S_a:

$$S_a = b_a \times c_a = 0.0212m/s^2$$

In the case of a rolling motion, the summation of all rolling moments (including the aircraft aerodynamic moment) is equal to the aircraft mass moment of inertia about the x-axis multiplied by the time rate of change of the roll rate (P):

$$\sum M_{cgx} = L_A - \Delta D \cdot y_D$$

Where,

$$L_A = 2\Delta L \cdot y_A = \frac{1}{2}\rho V_T^2 S C_l b = \overline{q} S C_l b$$

$$C_l = \text{Rolling Moment Coefficient} V_T = \text{Aircraft True Air Speed} S = \text{Wing Area} b = \text{Wing Span}$$

Drag Induced by Rolling
Aircraft rolling drag induced by the rolling speed may be modelled as:

$$D_R = (\Delta D_L + \Delta D_R) = \frac{1}{2}\rho V_R^2 S_{\text{TOT}} C_{D_R}$$

$$C_{D_R} = \text{Aircraft Drag Coefficient in Rolling} \quad (0.7 - 1.2)$$

$$S_{\text{TOT}} = (\text{Wing Area} + \text{Horizontal Tail Area} + \text{Vertical Tail Area}) = (S_W + S_{HT} + S_{VT})$$

$$V_R = \text{Rolling Linear Speed} = \text{Roll rate} \times y_d = (P \cdot y_d)$$

y_d = Average distance between the rolling drag center and the x-axis (i.e., aircraft center of gravity). Typically, 40% of the

The steady-state value for the roll rate (Pss) is obtained by considering the fact that when the aircraft is rolling with a constant roll rate, the aileron-generated aerodynamic rolling moment is equal to the moment of aircraft drag in the rolling motion:

$$L_A = \frac{1}{2}\rho(P \cdot y_D)^2 (S_W + S_{HT} + S_{VT}) C_{D_R} y_D$$

Solving for the steady-state Roll Rate (Pss) results in:

$$P_{SS} = \sqrt{\frac{2L_A}{\rho(S_W + S_{HT} + S_{VT}) C_{D_R} y_D^3}}$$

$$
\begin{aligned}
L_A &= \frac{1}{2}\rho V_T^2 S C_l b \\
&= 0.5 \times 1.225 \times (20^2) \times 0.42 \times 0.0854 \times 1.84 \\
&= 16.169 \\
P_{SS} &= \sqrt{\frac{2 \times 16.169}{1.225(0.42 + 0.069 + 0.04) \times 0.95 \times (0.368)^3}} \\
&= 32.466\,\mathrm{rad/s}
\end{aligned}
$$

Total Rolling Moment about CG can be calculated as,

$$L_A + \Delta D \cdot y_D = I_{xx}\dot{P}$$

$$\dot{P} d\phi = P dP$$

Bank angle can be calculated as,

$$\int_0^{\phi} \dot{P} d\phi = \int_0^{P_{SS}} P dP$$

$$\phi = \int_0^{P_{SS}} \frac{I_{xx} P}{L_A + \Delta D \cdot y_D} dP$$

$$\phi = \int_0^{P_{SS}} \frac{I_{xx} P}{\overline{q} S C_l b + \frac{1}{2}\rho (P y_D)^2 (S_W + S_{HT} + S_{VT}) C_{D_R} y_D} dP$$

$$\phi_1 = \left[\frac{I_{xx}}{(\rho y_D)^3 (S_W + S_{HT} + S_{VT}) C_{D_R}} \ln \left(\frac{P^2 + \frac{V^2 S C_l b}{(y_D)^3 (S_W + S_{HT} + S_{VT}) C_{D_R}}}{1} \right) \right]_0^{P_{SS}}$$

$$\phi_1 = \frac{I_{xx}}{(\rho y_D)^3 (S_W + S_{HT} + S_{VT}) C_{D_R}} \ln P_{SS}^2$$

10.3.2 Elevator Design

The primary function of an elevator is to ensure longitudinal stability by altering the $C_m 0$ of the aircraft by varying the lift generated by the tail. The major constraints for designing an elevator are the ability to meet the requirements during takeoff and trim. In other words, the elevator should produce enough pitching moment to meet the required angular acceleration required during takeoff within the maximum deflection limits in the upward and downward direction, while also being able to trim the aircraft within the limits specified.

The following requirements are also posed as the constraints for the elevator deflection.

$$\text{Takeoff pitch angular acceleration} = 10 \tag{10.6}$$

$$C_m = 0 \text{ (at cruise conditions)} \tag{10.7}$$

From [11], the initial values for the control surface to tail ratio for the chord, span and area are estimated. They are assumed as

$$\frac{b_e}{b_t} = 0.88 \tag{10.8}$$

$$\frac{S_e}{S_t} = 0.32 \tag{10.9}$$

$$\frac{C_e}{C_t} = 0.25 \tag{10.10}$$

Using these initial values, equations to satisfy the first condition to be satisfied is computed, which is required takeoff pitch angular acceleration of 10 rad^2/s

The following equations are used to find the aerodynamic forces and moments acting on the Aircraft during takeoff.

$$L_h = \frac{1}{2}\rho V_R^2 C_{L_h} S_h \tag{10.11}$$

$$L_{wf} = \frac{1}{2}\rho V_R^2 C_{L_{TO}} S_{ref} \tag{10.12}$$

$$D_{TO} = \frac{1}{2}\rho V_R^2 C_{D_{TO}} S_{ref} \tag{10.13}$$

$$M_{ac_{wf}} = \frac{1}{2}\rho V_R^2 C_{m_{ac_{wf}}} S_{ref}\overline{C} \tag{10.14}$$

The moments that these forces contribute to during takeoff is found using the following formulae

$$M_W = W(x_{mg} - x_{cg}) \tag{10.15}$$

$$M_D = D(Z_D - Z_{mg}) \tag{10.16}$$

$$M_T = T(Z_T - Z_{mg}) \tag{10.17}$$

$$M_{L_{wf}} = L_{wf}(x_{mg} - x_{ac_{wf}}) \tag{10.18}$$

$$M_{L_h} = L_h(x_{ach} - x_{mg}) \tag{10.19}$$

where all subscript "mg" denotes the main landing gear as we are calculating the moment about the main landing gear. Applying the moment balance at main landing gear, we get

$$\Sigma M_{mg} = I_{yy}\ddot{\theta} = -W(x_{mg} - x_{cg}) + D(Z_D - Z_{mg}) - T(Z_T - Z_{mg}) + L_{wf}(x_{mg} - x_{ac_{wf}}) + M_{acwf} - L_h(x_{ac_h} - x_{mg}) \tag{10.20}$$

From (10.20), I_{yy} denotes the moment of inertia of the UAV about an axis passing through both of the wheels of the main landing gear. This equation is used to find the amount of lift that the tail is expected to produce to get the required flight conditions of takeoff pitch angular acceleration.

$$L_h = \frac{[-W(x_{mg} - x_{cg}) + D(Z_D - Z_{mg}) - T(Z_T - Z_{mg}) + L_{wf}(x_{mg} - x_{ac_{wf}}) + M_{acwf} - I_{yy}\ddot{\theta}]}{x_{ac_h} - x_{mg}} \tag{10.21}$$

The moment of inertia I_{yy} about the main landing is found computationally using Fusion 360 tool. Substituting all the known values in the equation, we get the Lift required to be produced by the lift

$$L_h = -6.52N$$

Using this L_h we find the required Coefficient of lift for tail using the following equation

$$C_{L_h} = \frac{2 * L_h}{\rho V_R^2 S_{tail}} \tag{10.22}$$

From this equation, we get our required C_{L_h} as

$$C_{L_h} = -0.672$$

C_{L_h} varies with elevator deflection δ_e as the following equation

$$C_{L_h} = C_{L_{\alpha_h}}\alpha_h + C_{L_{\alpha_h}}\tau_e\delta_E = C_{L_{\alpha_h}}(\alpha_h + \tau_e\delta_E) \text{ Where} \tag{10.23}$$

$$\alpha_h = \alpha + i_h - \varepsilon \tag{10.24}$$

Using these two equations to find the value of δ_E, we get

$$\delta_E = -39.90°$$

Similarly finding the Moment balance at CG for trim Conditions (Finding moments of all the forces about CG of the UAV and equating it to 0), we get

$$\delta_E = -14.62°$$

As we can see, although the elevator can enable the tail to produce required lift during takeoff, it can not satisfy the conditions during takeoff. Hence the value of τ_e is altered and the calculations are redone until the final values of required elevator deflections δ_E fit within the specified limits

The value of τ_E is iterated over the set of values from 0.25 to 0.4 with step value of 0.05. The value of τ_E for which the elevator meets the requirement is 0.4

For $\tau_E = 0.4$, we get δ_E during takeoff $= -24.6°$ and during cruise as $-9.14°$

10.3.3 Rudder Design

Design of a Rudder is a complicated process when compared to other control surfaces because any deflection of rudder causes moment in both longitudinal and lateral axis and because of this, Rudder design always goes hand in hand with the aileron design. During flight, a rudder must be able to satisfy all of the following conditions.

1. Asymmetric Thrust - Should be able to fly without any heading angle even after one engine fails
2. Crosswind Landing - Must maintain alignment with the runway even when there is a crosswind
3. Spin Recovery - If it is a spinnable aircraft, should be able to recover from the spin
4. Coordinated turn - Turning without any sideslip angle
5. Adverse Yaw - Must be able to overcome adverse yaw produced by the ailerons

Although all Aircraft's should satisfy all of the above mentioned criteria for it to fly properly, we can design it to satisfy only the most critical flight condition depending on the type of the aircraft. From [11], we can say that the most critical design requirement for a UAV would be making a coordinated turn and Adverse Yaw.

A coordinated level turn happens when the net force along the body fitted y-axis becomes 0. By Newton's laws of motion, the equations the forces and moment that act on the y axis of the aircraft are

$$F_{A_{y_t}} = F_C - W sin\phi \tag{10.25}$$
$$L_{A_{y_t}} = (I_{zz} - I_{xx})R_1 Q_1 \tag{10.26}$$
$$N_{A_t} = I_{xz} R_1 Q_1 \tag{10.27}$$

Where R_1 represents Yaw rate and Q_1 represents pitch rate. For preliminary design, the following dimensions are assumed for the rudder

$$\frac{S_r}{S_{Vt}} = 0.15 \tag{10.28}$$
$$\frac{b_r}{b_{Vt}} = 0.7 \tag{10.29}$$
$$\frac{C_r}{C_{Vt}} = 0.15 \tag{10.30}$$

The following equations are the equations of forces and moments in the lateral-directional directions during a turn

$$F_C = m\frac{U_1^2}{R_1} \tag{10.31}$$
$$F_{A_t} = \frac{1}{2}\rho u_1^2 S\left(C_{Y_\beta}\beta + C_{y_r}\frac{R_1 b}{2U_1} + C_{y_{\delta_A}}\delta_A + C_{y_{\delta_R}}\delta_R\right) \tag{10.32}$$
$$L_{A_t} = \frac{1}{2}\rho u_1^2 Sb\left(C_{l_\beta}\beta + C_{l_r}\frac{R_1 b}{2U_1} + C_{l_{\delta_A}}\delta_A + C_{l_{\delta_R}}\delta_R\right) \tag{10.33}$$
$$N_{A_t} = \frac{1}{2}\rho u_1^2 Sb\left(C_{n_\beta}\beta + C_{n_r}\frac{R_1 b}{2U_1} + C_{n_{\delta_A}}\delta_A + C_{n_{\delta_R}}\delta_R\right) \tag{10.34}$$

Where all C_x are stability and control derivatives that are estimated using the XFLR5 tool. Similar to elevator design, using the initial values, the equations (10.27) and (10.34) are solved simultaneously to get rudder deflections under the limit conditions of aileron deflection. If the deflections fall under the ideal limits, we can take the solution but if it doesn't the span and the chord ratios are varied until it satisfies the required conditions. The stability derivatives are computed as a function of tail dimensions and airfoil characteristics and aircraft parameters like tail volume ratio, and numerically iterated until the desired conditions are met. After the iterations, the following values have seemed to produce the optimum conditions to satisfy the design requirements of the rudder.

- $\frac{S_r}{S_{Vt}} = 0.24$
- $\frac{b_r}{b_{Vt}} = 0.85$
- $\frac{C_r}{C_{Vt}} = 0.28$

Refer Appendix A.6 for the code used to finalise control surface sizing.
The final Dimensions (as ratios to the respective lifting surface) are given in the table below

Control Surface	Chord Ratio	Span Ratio	Area Ratio
Elevator	0.4	0.88	0.32
Rudder	0.28	0.85	0.24
Ailerons	0.2	0.25	0.05

Table 10.1: Final Dimensions of the Control Surfaces

For flaps, we keep generic sizes which is:

- chord ratio of 0.2 - 0.3.
- span ratio of 0.4.

Chapter 11

Performance Analysis

11.1 Drag Estimation

Drag (D) for an aircraft is a summation of parasitic drag (C_{D_0}) and lift induced drag.
We can write the Drag Coefficient, D as:

$$C_D = C_{D_0} + KC_L^2$$

where, C_{D_0} is the parasitic drag component and KC_L^2 is the lift induced drag component.

11.1.1 Parasitic Drag

Two methods for the estimation of the parasite drag are given in [10]:

1. Equivalent Skin-Friction Method- This method is sufficient for a well-built professional UAV.
2. Component Buildup Method - The component buildup method estimates the subsonic parasite drag of each component of the aircraft.

We are following the component buildup method for parasitic drag estimation.

$$C_{D_0} = \frac{\sum_c (C_{f_c} FF_c Q_c S_{\text{wet}_c})}{S_{\text{ref}}} + C_{D_{\text{misc}}} + C_{D_{L\&P}}$$

where,

- C_f is the flat-plate skin-friction drag coefficient.
- FF is the form factor that estimates the pressure drag due to viscous separation.
- Q is the interference factor that estimates the interference effects on the component drag.
- $C_{D_{\text{misc}}}$ accounts for the miscellaneous drags for special features of an aircraft such as flaps, un-retracted landing gear, an upswept aft fuselage, and base area.
- $C_{D_{L\&P}}$ accounts for the drag contributions due to leakages and protuberances.

Skin-friction drag coefficient calculation

[10] gives the formula for calculating C_f for laminar and turbulent flows.

$$C_{f,laminar} = \frac{1.328}{\sqrt{Re}}$$

$$C_{f,Turbulent} = \frac{0.455}{(log_{10}Re)^{2.58}(1 + 0.144M^2)^{0.65}}$$

where,

- Re is the Reynolds number of the flow and is given by

$$Re = \frac{\rho v l}{\mu}$$

  ρ is the density of fluid, v is the velocity, l is the length scale. For a fuselage, l is the total length. For a wing or tail, l is approximated by the mean aerodynamic chord length.

- M is the mach number of the flow and is given by
  - $M = \frac{v}{a}$

  v is the velocity of aircraft and a is the speed of sound.

From our mission statement, we are aiming to cruise at 100m altitude. The conditions of flight are:

- Cruise velocity, v = 20 m/s.
- Density of air, $\rho = 1.2 Kg/m^3$.
- Temperature at an altitude of 100m is taken to be 25^{circ}.
- Speed of sound = $\sqrt{\gamma RT} \approx 346 m/s$.
- Dynamic viscosity of air, $\mu = 1.85 \times 10^{-5} Kg/m.s$.
- M = $\frac{20}{346}$ = 0.058.

[10] Mentions that a good estimate for laminar flow over the surface would be 25% for fuselage and 40% for wing and tails.
We can determine the overall C_f by adding up the laminar and turbulent part along woith their fractions.

$$C_f = f_{laminar} C_{f,laminar} + (1 - f_{laminar}) C_{f,turbulent}$$

Table showing the values calculated is as follows.

Component	**Length Scale (m)**	**Re** ($\times \mathbf{10^5}$)	$\mathbf{C_{f,laminar}}$	$\mathbf{C_{f,turbulent}}$	$\mathbf{K_{laminar}}$	$\mathbf{C_f}$
Wing	0.23	0.2984	0.0024	0.00566	0.4	0.0044
Fuselage	1.69	2.1924	0.000897	0.0038	0.25	0.0033
Horizontal Tail	0.1137	0.1475	0.00346	0.0066	04	0.0053
Vertical Tail	0.1198	0.155	0.003369	0.0065	0.4	0.0052

Form Factor(FF) Calculation

The calculated flat-plate skin-friction coefficient must be adjusted upwards to take into account the pressure drag caused by flow separation - Prandtl's solution to d'Alembert's paradox. This is done with empirical "form factors" that have been derived from theoretical and empirical considerations [10].
For Wing and Tail,

$$FF = \left[1 + \frac{0.6}{(x/c)_m}\left(\frac{t}{c}\right) + 100\left(\frac{t}{c}\right)^4\right]\left[1.34M^{0.18}(\cos\Lambda_m)^{0.28}\right]$$

where,

- the term $(\frac{x}{c})_m$ is the chord-wise location of the airfoil maximum thickness point. For most low-speed airfoils, this is at about 0.3 of the chord.
- $(\frac{t}{c})$ for NACA2412 and NACA0012 airfoils are 0.12.
- The sweep of the maximum-thickness line, Λ_m is zero for a rectangular wing or tail.

Substituting values, we get $FF_{wing} = FF_{HT} = FF_{VT} = 1.012$. For fuselage,

$$FF = (0.9 + \frac{5}{f^{1.5}} + \frac{f}{400})$$

where,

$$f = \frac{l}{d} = \frac{l}{\sqrt{(4/\pi)A_{max}}}$$

with l, d and A_{max} being the length, effective diameter and maximum area of the fuselage.
Substituting values, we get f = 9.88 and hence, $FF_{fus} = 1.0857$.
Since, we are taking a rectangular cross-section of fuselage, we should take 30-40% increment in the FF of fuselage.
Hence, $FF_{fus} = 1.3 \times 1.0857 = 1.4114$.

Component	FF
Wing	1.012
Fuselage	1.4114
Horizontal Tail	1.012
Vertical Tail	1.012

Component Interference Drag

Parasitic drag is increased by the mutual interference between components. This is a catch-all phrase for the various ways that two components, brought together, will have more drag than the sum of their separate drags as per [10]. Wherever two components intersect each other, like the wing and fuselage or the vertical and horizontal tails, their boundary layers interact. As a result, the boundary layer is thicker, causing more drag.

- For high-wing, the interference will be negligible so that the Q factor ≈ 1.
- Fuselage has a negligible factor, so Q ≈ 1.
- For a conventional tail, 4-5% of interference factor can be assumed. So, Q ≈ 1.04.

Wetted Area Calculation

The wetted areas of Wing, Fuselage, Horizontal Tail and Vertical Tail are obtained from the CAD model and tabulated as follows:

Component	$\mathbf{S_{wet}}$ $\mathbf{(m^2)}$
Wing	0.861
Fuselage	0.6295
Horizontal Tail	0.1405
Vertical Tail	0.07512

Substituting values. we get $\frac{\sum_c (C_{f_c} FF_c Q_c S_{\text{wet}_c})}{S_{\text{ref}}} = 0.0185$.

11.1.2 Miscellaneous Drags

We consider the landing gear drag and the drag due to upsweep for miscellaneous drags.
For upsweep, [10] mentions the formula for finding the drag area,

$$\frac{D}{q_{upsweep}} = 3.83u^{2.5}A_{max}$$

where, u is the upsweep angle in radians = 0.1086 rad, A_{max} is the maximum fusleage area = 0.0229 m^2. We get $\frac{D}{q_{upsweep}} = 0.00034$. We get C_D due to upsweep by considering $\frac{D}{q_{upsweep}S_{ref}} = 0.0008$.
For tires, we take $C_{D_\pi} = \frac{D/q}{FrontalArea} = 0.15$ from [10]. Frontal area for 4 tires = 0.004752 m^2. From [10], we take $C_{D_\pi} = 0.15$ for tires. Hence, we get the C_{D_0} due to tires,

$$C_{D_{0,tires}} = \frac{0.15 \times 0.004752}{0.42} = 0.00169$$

For struts, the C_{D_π} is taken to be 0.1 from [10] and frontal area = 0.018 m^2. We get,

$$C_{D_{0,struts}} = \frac{0.05 \times 0.018}{0.42} = 0.0021$$

11.1.3 Leakage and Protuberance contribution

Leakage and protuberances add drag that is difficult to predict by any method. Leakage drag is due to the tendency of an aircraft to "inhale" through holes and gaps in high-pressure zones and "exhale" into the low pressure zones. Since, we are not considering an air-breathing system, we can write take $C_{L\&P} = 0$.

Summing up, we get

$$C_{D_{misc}} = C_{D_{0,upsweep}} + C_{D_{0,tires}} + C_{D_{0,struts}} = 0.0008 + 0.00169 + 0.0021 = 0.00459$$

Considering , we get,

$$C_{D_0} = 0.0185 + 0.00459 = 0.0231$$

11.1.4 Drag Polar

The Drag polar is a plot of C_D vs C_L. They are related by,

$$C_D = C_{D_0} + KC_L^2$$

where,

- $C_{D_0} = 0.0231$ as found.
- $K = \frac{1}{\pi e AR} = \frac{1}{\pi \times 0.816 \times 8} = 0.04876$ as found earlier.

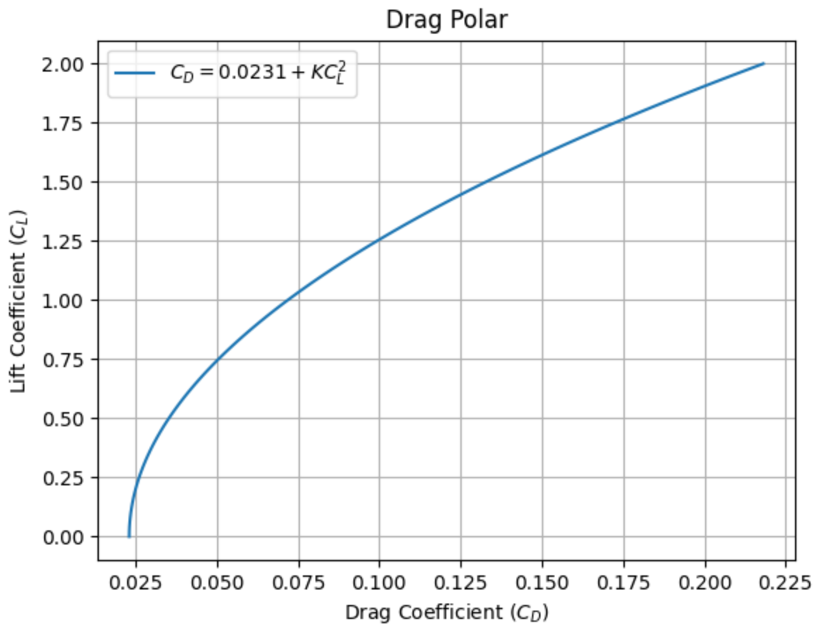

Figure 11.1: Drag Polar, C_L vs C_D

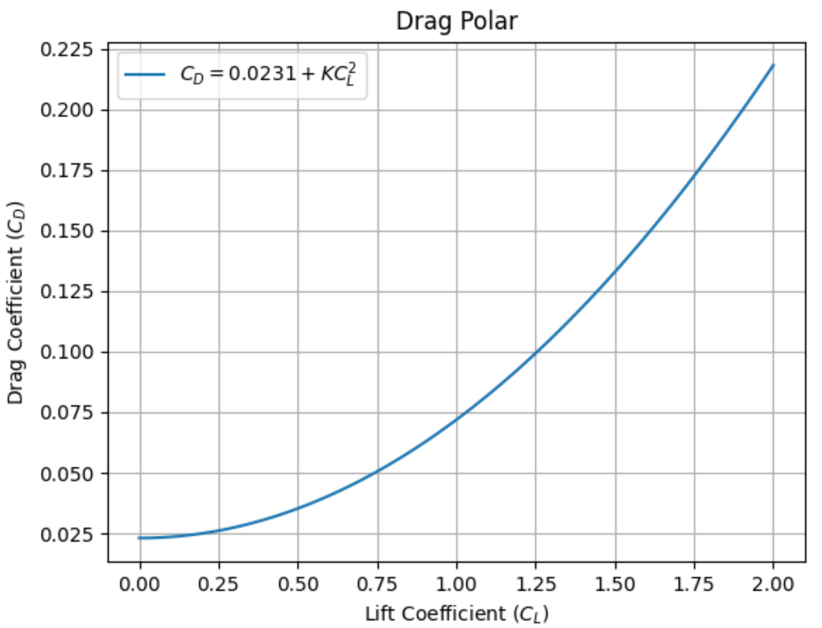

Figure 11.2: Drag Polar, C_D vs C_L

11.2 Power Requirement

The thrust generated by the aircraft is equal to the drag experienced in a non-accelerated flight situation. The power required (P_R) is defined as the power that is required to produce the thrust required to sustain a non-accelerated flight. That is,

$$P_R = TV = DV$$

where T is Thrust, D is Drag and V is velocity.
The Drag(D) is given by,

$$D = \frac{1}{2}\rho V^2 S C_D$$

where ρ is the density, S is the wing planform area and C_D is drag coefficient.

We know,

$$C_D = C_{Do} + KC_L^2$$

$$C_D = C_{Do} + K\left(\frac{2w}{\rho V^2 S}\right)^2$$

$$C_D = 0.0231 + 0.04876\left(\frac{2 \times 5.47 \times 9.81}{1.2 \times V^2 \times 0.42}\right)^2$$

$$C_D = 0.0231 + \frac{2206.43}{V^4}$$

Drag amounts to

$$D = 0.5 \times 1.2 \times V^2 \times 0.42 \times C_D$$

$$D = 0.0058V^2 + \frac{556.02}{V^2}$$

Now, Power required is

$$P_R = DV = 0.0058V^3 + \frac{556.02}{V}$$

The Power Available(P_A) is the maximum power that is available to produce thrust. P_A stays constant in an airplane powered by a propeller. The P_A for our UAV is 333 Watts from the battery. The following plot P_R and P_A with respect to velocity.
We can plot the Power vs Velocity plot.

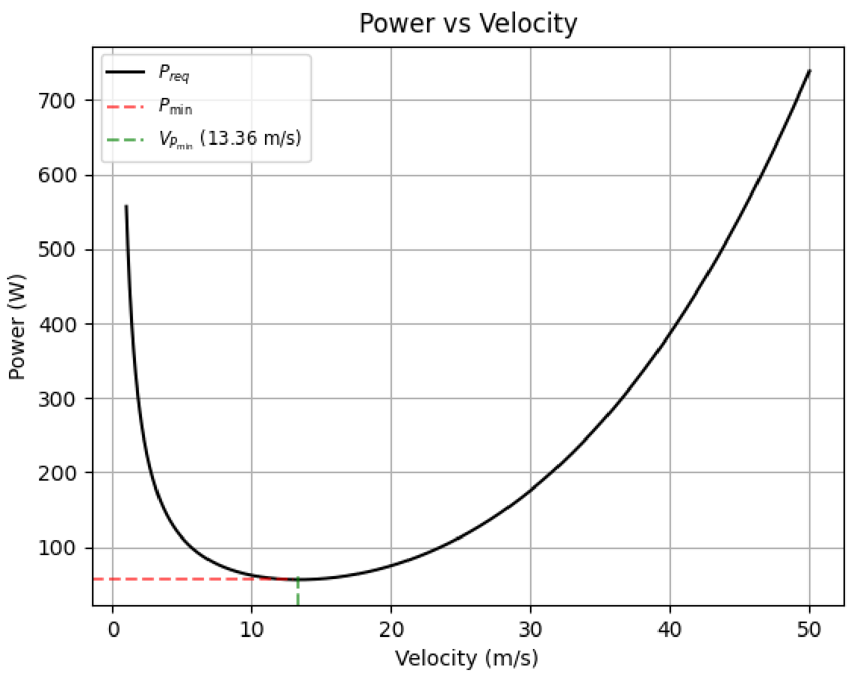

Figure 11.3

11.3 Climb Rate

The rate of climb (R/C) is the vertical velocity of the aircraft during its climb. It is the result of multiplying the aircraft's velocity (V) by the climb angle's sine (γ).i.e,

$$R/C = V sin(\gamma)$$

From the governing equations of aircraft during its climb, we know that

$$T = D + W sin(\gamma)$$

$$\implies \frac{TV - DV}{W} = V sin(\gamma)$$

Since TV is P_A, we can write it as

$$R/C = \frac{P_A - \frac{1}{2}\rho V^3 S C_D}{W}$$

The following plot shows the variation of R/C with respect to the velocity.

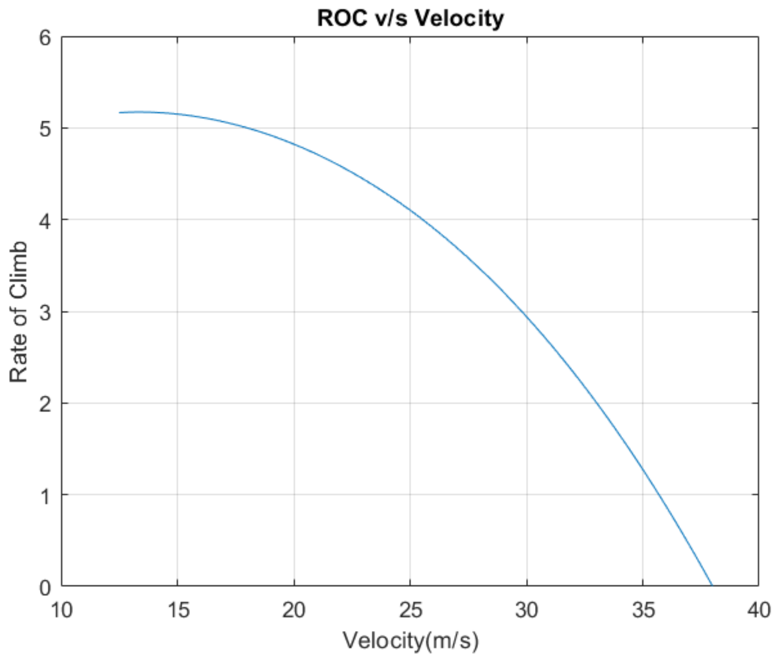

Figure 11.4: Rate of climb v/s Velocity

11.4 Climb Angle

The Climb angle(γ) is angle made by velocity vector with the horizontal axis. As per above R/C equation, we can write γ as inverse sine of rate of climb by Velocity.

$$\gamma = sin^{-1} \frac{R/C}{V}$$

$$\gamma = sin^{-1} \left[\frac{P_A - \frac{1}{2}\rho V^3 S C_D}{WV} \right]$$

The below plot shows variation of γ with respect to the Velocity.

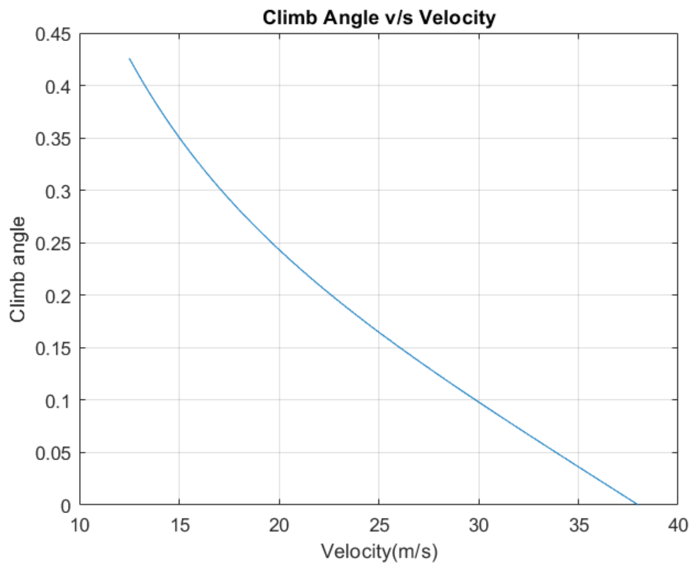

Figure 11.5: Climb angle v/s Velocity

11.5 V-n Diagram

A graph of speed versus load factor (or V-n diagram) is a way of showing the limits of an aircraft's performance. It shows how much load factor can be safely achieved at different airspeeds. The V-n diagram is hence, a plot of load factor vs velocity with aerodynamic and structural constraints imposed.
The maximum value n is called the limit load factor. [10] gives the range of positive and negative limit load factors.
We take $n_{positive} = 2.5$ and $n_{negative} = -1$. We know that,

$$n = \frac{1}{2}\rho_\infty V_\infty^2 \frac{C_{L_{max}}}{W/S}$$

The maximum velocity allowed is the red-line speed for the airplane. It is 1.2 times the maximum velocity of any phase as given in [10]. The red-line speed for us would be 30 m/s from a max cruise speed of 25 m/s. We find the maximum positive C_L = 1.43 and maximum negative C_L = -1.24 from the lift curve plot based on our airfoil for wing and then finding them for the wing.

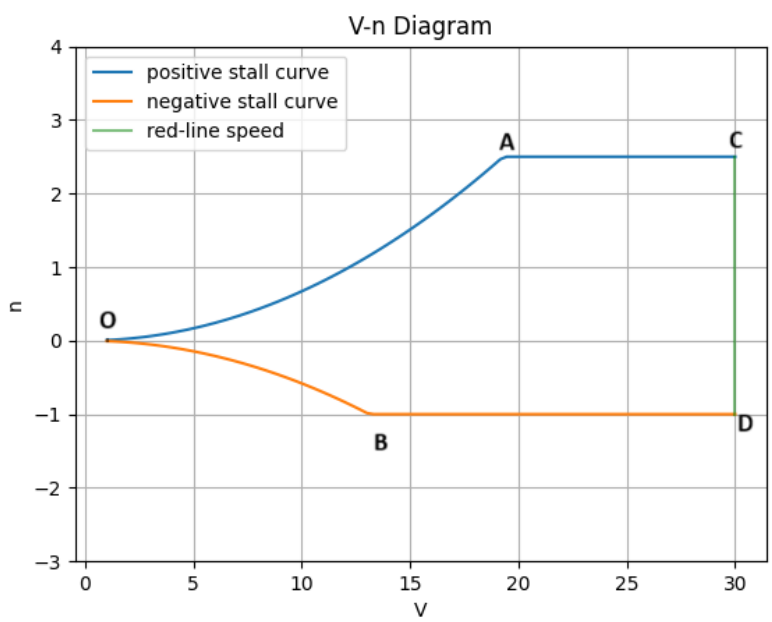

Figure 11.6: V-n diagram for the UAV

In the above V-n diagram,

- Part O-A is the positive stall curve, O-B is the negative stall curve.
- A-C represents the positive limit factor and B-D represents the negative stall curve.
- C-D represents the red-line speed limit of the UAV = 30m/s and this speed should never be exceeded.
- A is the maneuver point and the velocity corresponding to the corner point is the corner speed given by

$$V_{corner} = \sqrt{\frac{2n_{positive}}{\rho_{\infty} C_{L_{max}}} \frac{W}{S}} = 19.29 m/s.$$

- The stall velocity can be calculated as,

$$V_{stall} = \sqrt{\frac{2W}{\rho S C_{L,max}}} = 12.203 m/s$$

Refer Appendix A.7, A.8, A.9, A.10 and A.11 for the codes used.

11.6 Range and Endurance

We can estimate the endurance as the maximum amount of time the UAV can stay in the air. This can be done by assuming the UAV is in cruise state for the whole of flight.
We know that the power available in the battery chosen is 11.1 × 10 VAh = 111 Wh. The power required for cruise phase was calculated to be 65.4 W. So we get the endurance from,

$$E = \frac{111}{65.4} = 1.69 \text{ hours}$$

Range, R can be estimated as the product of cruise speed and Endurance,

$$R = V_{cr}E = 20 \times 1.69 \times \frac{18}{5} = 121.68\text{Km}$$

11.7 Takeoff Performance

We need to ensure that the required speed for takeoff is reached at the end of ground roll so that we can confirm that takeoff can take place.
In Chapter 3, we found the acceleration for ground roll as 1.286 m/s^2. We know that the time of ground roll is taken to be 10 seconds. We need lift off speed to be just higher than the stall velocity.

$$V_{LO} = 1.286 \times 10 = 12.86 m/s$$

V_{stall} was found to be 12.203 m/s. Hence, we get that the speed at the end of ground roll is sufficient to takeoff.

Appendix A

Codes Used

A.1 First Weight Iteration

```python
#Python file
import matplotlib.pyplot as plt
import pandas as pd
import numpy as np
import math

W = [5.5,2.2,2,2.7,5.5,4,10,22]
EW = [2.4,1.82,1.72,1.7,2.8,2,4.4,14]
EWF = []
for i in range(len(W)):
    EWF.append(EW[i]/W[i])

log_EWF = np.log(EWF)
log_W = np.log(W)


coefficients = np.polyfit(log_W, log_EWF, 1)
line = np.polyval(coefficients, log_W)


slope = coefficients[0]
intercept = coefficients[1]

#print("L = ",slope)
#print("A = ",math.exp(intercept))


plt.plot([math.log(x) for x in W], [math.log(x) for x in EWF],'bo')
plt.plot(log_W, line, label='Fitted Line')
plt.ylabel('log(EW/W)')
plt.xlabel('log(W)')
plt.title('Logarithmic plot of EW/W against W')
plt.show()

A = 0.76
L = -0.16

W1=[]
Wexp=[]
x1=[]
for i in range(25):
    x1.append((i+1))
    Wexp.append(A*math.pow((i+1),L))
```

```
for num in W:
    Wl.append(math.log(num))
x=W
y=EWF

for weight in range(len(Wl)):
    y[weight] = math.exp(log_EWF[weight])
plt.plot(x,y,'o',x1,Wexp,'-')
plt.title("Empty weight fraction vs DTOW")
plt.legend(["data points","best fit curve"])
plt.xlabel("DTOW")
plt.ylabel("Empty Weight Fraction")
plt.show()

def cl(w0,v):
    return (w0*9.81)/(0.6*(v**2)*0.78)

def cd(cl):
    return 0.04 + 0.047*(cl**2)



df=pd.DataFrame(columns=["S.No","W0","We/W0","W0new"])
wp=1.15
wc = 0

w0old = 12

w0new = 0
i=1
A=0.76
L= -0.16
wef = A*(w0old**L)
#print("wef = ",wef)


s = 0.78
v_cr = 20
t_cr = 117*60
t_climb = 60

w0_curr = [w0old]
iter = [0]



while(True):
    df1=pd.DataFrame({"S.No":i,"W0":w0old,"We/W0":wef,"W0new":w0new},index
    =[0])
    if(i>2000):
        break

    v_climb = 4.27*(w0old**0.5)
    cl_old = cl(w0old,v_cr)
    cl_climb = cl(w0old,v_climb)


    D_cr = cd(cl_old)*0.6*(v_cr**2)*s
    D_climb = cd(cl_climb)*0.6*(v_climb**2)*s

    P_cr = D_cr*v_cr
```

```
    P_climb = D_climb*v_climb
    E_ground = (1.21 * ((w0old*9.81)**2))/(9.81*1.225*1.62*s)*2
    P_loiter = 1612.41*1.7 * (0.04 + 0.000522 * (w0old**2))

    #print("P_loiter =",P_loiter)
    #print("P_cruise = ", P_cr)
    #print("P_climb = ", P_climb)

    E_cr = P_cr * t_cr*0.1
    E_climb = P_climb*t_climb*2
    E_loiter = P_loiter*t_cr*0.9

    #print("E_climb = ", E_climb)
    #print("E_loiter = ", E_loiter)
    #print("E_cr = ",E_cr)
    #print("E_ground = ", E_ground)

    E = E_cr + E_loiter + E_ground + E_climb




    wb = 1.33*E*0.000278/250
    #print("wb/w0old = ",wb/w0old)

    w0new = (wp+wc)/(1-(wb/w0old)-wef)
    # print("Iteration ",i,"done")
    # print(wef,w0new,w0old)
    if(abs(w0new-w0old)<0.01):
        print("Solution Converged with W0 = ",w0new, " After ",i," Iterations")
        break
    else:
        w0old = w0new
        wef = A*pow(w0new,L)
    df=df._append(df1,ignore_index=True)
    i=i+1
    #print("wef = ",wef)
    #print("w0new = ",w0new)
    w0_curr.append(w0new)
    iter.append(i)


print("wb = ",wb)
print("wef = ",wef)
print("w0new = ",w0new)
print("wb/w0old = ",wb/w0old)

plt.plot(iter,w0_curr)
plt.xlabel("Number of iterations")
plt.ylabel("DTOW")
plt.title("DTOW vs  Number of iterations")
plt.show()

print("cl = ",cl(w0new,v_cr))
print("E",E)
print("wb/w0 = ",wb/w0new)
df.astype({"S.No":int})

print(df.dtypes)
df.to_latex("Weight_Calc.tex",index=False)
```

```
print(df)
```

A.2 $(\frac{L}{D})_{max}$ vs $\frac{T}{W}$

```
#Python file
import matplotlib.pyplot as plt
import numpy as np
import math


b = [1.96 , 1, 3, 1.63, 2.6, 1.2]
l = [1.07, 0.66, 0.74, 1.2, 1.25, 0.91]
s_wing = [0.43,0.14,0.68,0.324,0.43,0.22]
a_top = [0.144, 0.063, 0.482,  0.111, 0.11, 0.045]
a_side = [0.168, 0.054,0.028, 0.128, 0.103, 0.103 ]
s_tail = [0.127,0.03,0.2,0.082,0.1,0.03]
v_cr = [20, 15, 19, 18.05, 15.28, 25.55]


def s_fus_wet_calc(a1,a2):
    return 1.7*(a1+a2)

def s_wet_wing_tail_calc(a1,a2):
    return 2.003*(a1+a2)

def AR_calc(b,s):
    return b**2/s

def R_calc(v,l):
    return (1.2 * v*l/1.849)*(10**5)

def e_calc(ar):
    return 1/(1.05 + 0.007*math.pi*ar)

def k_calc(e,ar):
    return 1/(math.pi*e*ar)

def M_calc(v):
    return v/346.03

def cf_calc(R,M):
    return 0.455/((((math.log10(R)))**2.58)*((1 + 0.144 * (M**2)))**0.65)

def cd0_calc(cf,s_wet,s):
    return cf*s_wet/s

def l_by_d_calc(cd0,k):
    return math.sqrt(1/(4*cd0*k))

s_fus_wet = []
s_wet_wing_tail = []
AR = []
R = []
e = []
k = []
M = []
cf = []
cd0 = []
```

```
l_by_d = []
s_wet = []
AR_wet = []
Swet_by_S = []

for i in range(len(b)):

    s_fus_wet_temp = (s_fus_wet_calc(a_top[i],a_side[i]))
    s_fus_wet.append(s_fus_wet_temp)

    s_wet_wing_tail_temp = (s_wet_wing_tail_calc(s_tail[i],s_wing[i]))
    s_wet_wing_tail.append(s_wet_wing_tail_temp)

    s_wet_temp = s_wet_wing_tail_temp + s_fus_wet_temp
    s_wet.append(s_wet_temp)

    AR_temp = (AR_calc(b[i],s_wing[i]))
    AR.append(AR_temp)

    AR_wet_temp = ((b[i])**2)/s_wet_temp
    AR_wet.append(AR_wet_temp)

    R_temp = (R_calc(v_cr[i],l[i]))
    R.append(R_temp)

    M_temp = (M_calc(v_cr[i]))
    M.append(M_temp)

    e_temp = e_calc(AR_temp)
    e.append(e_temp)

    k_temp = k_calc(e_temp,AR_temp)
    k.append(k_temp)

    cf_temp = cf_calc(R_temp,M_temp)
    cf.append(cf_temp)

    cd0_temp = cd0_calc(cf_temp,s_wet_temp,s_wing[i])
    cd0.append(cd0_temp)

    Swet_by_S_temp = s_wet_temp/s_wing[i]
    Swet_by_S.append(Swet_by_S_temp)

    l_by_d_temp = l_by_d_calc(cd0_temp,k_temp)
    l_by_d.append(l_by_d_temp)


print("AR = ",AR)
print("AR_wet = ",AR_wet)
print("k = ",k)
print("e = ",e)
print("cd0 = ",cd0)
print("S_wing = ",s_wing)
print("S_wet = ",s_wet)
print("l/d = ",l_by_d)
print("Swet/S = ", Swet_by_S)
print("S")

sqr_AR_wet = []
p = 0

```

```
for i in range(len(b)):
    sqr_AR_wet.append(math.sqrt(AR_wet[i]))
    p += Swet_by_S[i]

Swet_by_S_avg = p/len(b)

print("S_wet/S Avg = ",Swet_by_S_avg)




# Convert numbers to strings and format as LaTeX-style expressions
sq_ar_labels = [f'$\\sqrt{{{num:.2f}}}$' for num in sqr_AR_wet]
l_d_labels = [f'$(L/D)_{{max}} = {num:.2f}$' for num in l_by_d]

# Linear regression
regression_coefficients = np.polyfit(sqr_AR_wet, l_by_d, 1)
regression_line = np.polyval(regression_coefficients, sqr_AR_wet)

# Print the equation of the line
equation_of_line = f"$(L/D)_{{max}} = {regression_coefficients[0]:.2f} \\
   cdot \\sqrt{{AR_{{wet}}}} + {regression_coefficients[1]:.2f}$"
print("Equation of the line:", equation_of_line)

plt.plot(sqr_AR_wet, l_by_d, 'o', label='Data points')
plt.plot(sqr_AR_wet, regression_line, label='Linear Regression')

plt.ylabel("$(L/D)_{max}$")
plt.xlabel("$\\sqrt{AR_{{wet}}}$")  # Update the label
plt.title("$(L/D)_{max}$ vs $\\sqrt{AR_{{wet}}}$")  # Update the title
plt.grid()
plt.text(1.9, 17, equation_of_line, fontsize=10, verticalalignment='bottom',
    horizontalalignment='right')
plt.legend()
plt.show()


AR_our = 8
b_our = 2.5
s_wing_our = 0.78
v_our = 20

s_wet_our = 3.674 * s_wing_our
AR_wet_our = b_our**2/s_wet_our

print("AR_wet_our = ", AR_wet_our)
l_by_d_our = 9.89*math.sqrt(AR_wet_our) + 3.22

print("l/d _ our = ",l_by_d_our)

T_W = 1/l_by_d_our

T_cr = T_W * 7.826 * 9.81

print("T_W = ",T_W)
print("T_cr= ", T_cr)

P_cr_our = T_cr * v_our
print("P_cr_our = ", P_cr_our)
```

A.3 Second Weight Estimation

```python
#Python file
import matplotlib.pyplot as plt
import pandas as pd
import numpy as np
import math


df=pd.DataFrame(columns=["S.No","W0","We/W0","W0new"])
i = 1
wp = 0.8
wb = 1.25
A=0.76
L= -0.16
w0old = 15
w0new = 0
w0_curr = [w0old]
wef = A*(w0old**L)
iter = [0]

while(True):
    df1=pd.DataFrame({"S.No":i,"W0":w0old,"We/W0":wef,"W0new":w0new},index
    =[0])
    if(i>2000):
        break

    w0new = (wp + wb) / (1 - wef)

    if(abs(w0new-w0old)<0.0001):
        print("Solution Converged with W0 = ",w0new, " After ",i,"
    Iterations")
        break
    else:
        w0old = w0new
        wef = A*pow(w0new,L)
    df=df._append(df1,ignore_index=True)
    i=i+1
    w0_curr.append(w0new)
    iter.append(i)
    #print("wef = ",wef)

#After adding 10% tolerance
print("W0new= ",w0new*1.1)
plt.plot(iter,w0_curr)
plt.xlabel("Number of iterations")
plt.ylabel("DTOW")
plt.title("DTOW vs  Number of iterations")
plt.grid()
plt.show()
```

A.4 Fuselage Length

```python
#Python file
import numpy as np
import matplotlib.pyplot as plt
from scipy.optimize import curve_fit

# Data
```

```python
l = np.array([1.11, 2.2, 2.25, 1.95, 1.25, 1.3, 1.11])
w0 = np.array([2.2, 10.7, 7, 8, 1.7, 5.5, 4])

# Define the function
def model(w0, a, c):
    return a * w0**c

# Perform the fitting
popt, pcov = curve_fit(model, w0, l)

a_fit, c_fit = popt

# Generate points for the fitted curve
w0_fit = np.linspace(min(w0), max(w0), 100)
l_fit = model(w0_fit, a_fit, c_fit)

# Plotting
plt.figure(figsize=(12, 6))

# Normal plot
plt.subplot(1, 2, 1)
plt.scatter(w0, l, label='Data')
plt.plot(w0_fit, l_fit, 'r-', label=f'Fit: a={a_fit:.2f}, c={c_fit:.2f}')
plt.xlabel('w0')
plt.ylabel('l')
plt.title("Fuselage lenngth vs MTOW")
plt.legend()

# Log-log plot
plt.subplot(1, 2, 2)
plt.loglog(w0, l, 'bo', label='Data')
plt.loglog(w0_fit, l_fit, 'r-', label=f'Fit: a={a_fit:.2f}, c={c_fit:.2f}')
plt.xlabel('log(w0)')
plt.ylabel('log(l)')
plt.title('Log-Log Plot')
plt.legend()

plt.tight_layout()
plt.show()

print(f"The values of a and c are: a = {a_fit:.2f}, c = {c_fit:.2f}")
L_our = a_fit*(5.47**c_fit)
print("L_our: ",L_our)
```

A.5 Fuselage Height and Width

```python
#Python file
import numpy as np
import matplotlib.pyplot as plt

# Input data
l = np.array([1.11, 2.2, 1.95, 2.25])
w = np.array([0.13, 0.27, 0.14, 0.15])
h = np.array([0.095, 0.084, 0.22, 0.16])

# Perform linear regression for width
coefficients_w = np.polyfit(l, w, 1)
poly_fit_w = np.poly1d(coefficients_w)

```

```
# Perform linear regression for height
coefficients_h = np.polyfit(l, h, 1)
poly_fit_h = np.poly1d(coefficients_h)

# Plotting for width
plt.figure(figsize=(8, 6))
plt.scatter(l, w, label='Width Data')
plt.plot(l, poly_fit_w(l), color='red', label='Linear Regression (Width)')
plt.xlabel('Length')
plt.ylabel('Width')
plt.title('Fuselage width vs length')
plt.legend()
# Annotate the equation of the linear regression line for width
equation_w = f'Width = {coefficients_w[0]:.2f} * Length + {coefficients_w
    [1]:.2f}'
plt.text(1.2, 0.2, equation_w, fontsize=10, color='black')
plt.grid(True)
plt.show()

# Plotting for height
plt.figure(figsize=(8, 6))
plt.scatter(l, h, label='Height Data')
plt.plot(l, poly_fit_h(l), color='green', label='Linear Regression (Height)'
    )
plt.xlabel('Length')
plt.ylabel('Height')
plt.title('Fuselage height vs length')
plt.legend()
# Annotate the equation of the linear regression line for height
equation_h = f'Height = {coefficients_h[0]:.2f} * Length + {coefficients_h
    [1]:.2f}'
plt.text(1.2, 0.2, equation_h, fontsize=10, color='black')
plt.grid(True)
plt.show()

# Calculate predicted width and height for l = 1.69
l_value = 1.69
w_predicted = poly_fit_w(l_value)
h_predicted = poly_fit_h(l_value)
print(f'Predicted width for l = {l_value}: {w_predicted}')
print(f'Predicted height for l = {l_value}: {h_predicted}')
```

A.6 Control Surface Sizing

```
#Python file
# %%
import matplotlib.pyplot as plt
import numpy as np

# %%
clalphaw = 0.0812

# %%
Clto = 1.4
Cdto = 0.04 + 0.048*Clto**2
Swing = 0.42
Lwf = 0.5*1.225*(15.38/1.1)**2*Clto*Swing
print(Lwf)
D = 0.5*1.225*(15.38/1.1)**2*Cdto*Swing
```

```
Cmacw = -0.05
Cwing = 0.23
Macwing = 0.5*1.225 * (15.38/1.1)**2 * Cmacw * Cwing

# %%
CMalpha = -2.069
I = 1.075
topas = 0
bel = 0.85
maxdefp = 20
maxdefn = -25
tvrh = 0.685
tef = 0.9
clalphah= 0.071
btail = 0.606
taue = 0.4
Stail = 0.069
Swing= 0.42
Weight= 5.36 * 9.81
Mwl = (198 - 186)*0.01
Mdl = (95-62)*0.01
Mtl = (93-62)*0.01
Mlwl = 0.1
Mlhl = (-74 + 167)*0.01
T = 5.12
Lh = (D*Mdl -T *Mtl + Lwf*Mlwl + Macwing - I*topas)/Mlhl
print(Lh)




# %%
Clh = 2*Lh/(1.225 * 20 * 20 * Stail)

# %%
Clh

# %%
effangle = Clh/clalphah
effangle = effangle
alpha = Clto/clalphaw
angle = effangle - (-0.042 + 0.542*alpha)
print(angle)

# %%
(angle)/taue

# %%
```

A.7 Parasitic Drag Calculation

```
#Python file
import math
rho = 1.2
mu = 1.85 * (10**-5)
v = 20
l = [0.23, 1.69, 0.1137, 0.1198]        # wing, fus, ht, vt

Re = []
```

```
cf_lam = []
cf_tur = []
K_lam = [0.4, 0.25, 0.4, 0.4]
cf = []
Q = [1 , 1, 1.04, 1.04]
Swet = [0.861, 0.6295, 0.1405, 0.075115]
FF = [1.012, 1.4114, 1.012, 1.012]
Sref = 0.42
cd = 0

for i in range(len(l)):
    Re_temp = rho*v*l[i]/mu
    Re.append(Re_temp)
    cf_lam_temp = (1.328/math.sqrt(Re_temp))
    cf_tur_temp = 0.455/(math.log10(Re_temp)**2.58)
    cf_tur.append(cf_tur_temp)
    cf_lam.append(cf_lam_temp)
    cf_temp = cf_lam_temp*K_lam[i] + cf_tur_temp*(1 - K_lam[i])
    cf.append(cf_temp)
    cd += cf_temp*FF[i]*Q[i]*Swet[i]/Sref

print("Re: ", Re)
print("cf_lam: ", cf_lam)
print("cf_tur: ", cf_tur)
print("cf: ", cf)
print("cd: ", cd)
```

A.8 Drag Polar

```
#Python file
import matplotlib.pyplot as plt
import numpy as np
import math

K = 1/(math.pi*0.816*8)
cl = np.linspace(0, 2, 100)
cd = 0.0231 + K * cl**2

plt.plot(cd, cl, label=r'$C_D = 0.0231 + KC_L^2$')
plt.title("Drag Polar")
plt.ylabel('Lift Coefficient ($C_L$)')
plt.xlabel('Drag Coefficient ($C_D$)')
plt.legend()
plt.grid(True)
plt.show()

plt.plot(cl, cd, label=r'$C_D = 0.0231 + KC_L^2$')
plt.title("Drag Polar")
plt.xlabel('Lift Coefficient ($C_L$)')
plt.ylabel('Drag Coefficient ($C_D$)')
plt.legend()
plt.grid(True)
plt.show()
```

A.9 Power required curve

```
#Python file
import matplotlib.pyplot as plt
```

```
import numpy as np
import math

cd0 = 0.0231
K = 0.04876
v = np.linspace(1, 50, 1000)
cl = []
w = 5.47 * 9.81
rho = 1.2
S = 0.42
P = []

for i in range(len(v)):
    cl_temp = 2*w/(rho*(v[i]**2)*S)
    P_temp = (cd0 + K*(cl_temp**2))*v[i]*(0.5*rho*S*(v[i]**2))
    cl.append(cl_temp)
    P.append(P_temp)

plt.plot(v, P, color = 'black', label = '$P_{req}$')
plt.xlabel('Velocity (m/s)')
plt.ylabel('Power (W)')
plt.title('Power vs Velocity')
plt.grid(True)

# Draw a horizontal line at the minimum point
min_P = min(P)
min_index = P.index(min_P)
plt.axhline(y=min_P, xmin=0, xmax=v[min_index]/max(v), color='r', linestyle=
    '--', label='$P_{\min}$', alpha=0.6)

# Draw a vertical line at the corresponding velocity
min_velocity = v[min_index]
plt.axvline(x=min_velocity, ymin = 0, ymax=min_P / max(P) , color='g',
    linestyle='--', label=f'$V_{{P_{{\min}}}}$ ({min_velocity:.2f} m/s)',
    alpha=0.6)

plt.legend(loc='best', fontsize='small')
plt.show()
```

A.10 Rate of Climb and Climb angle

```
%Matlab file
v = (12.5:0.05:38);
p1 = 0.0058*(v.^3);
p2 = 556.02*(1./v);
P = p1 + p2;
W = 5.47 * 9.81;
RC = (333 - P)./W;
gamma = asin(RC./v);
plot(v,RC);

grid on;
title("ROC v/s Velocity");
xlabel("Velocity(m/s)");
ylabel("Rate of Climb");

plot(v,gamma);

grid on;
```

```
title("Climb Angle v/s Velocity");
xlabel("Velocity(m/s)");
ylabel("Climb angle");
```

A.11 V-n diagram

```
#Python file
import matplotlib.pyplot as plt
import numpy as np
import math

v = np.linspace(1, 30, 100)
T_w = 1/17.826
w = 5.47 * 9.81
rho = 1.2
S = 0.42
cl_max = 1.43
cl_min = 1.24
v_stall = math.sqrt(2*w/(rho*S*cl_max))
v_stall_min = math.sqrt(2*w/(rho*S*cl_min))
w_s = w/S
K = 1/(math.pi*0.816*8)
cd0 = 0.0231

print(v_stall)

n = []
n1 = []

for i in range(len(v)):
    n_temp = 0.5*1.2*v[i]**2*cl_max/w_s
    n_temp = min(n_temp, 2.5)
    n.append(n_temp)
    n_min_temp = -0.5*1.2*v[i]**2*cl_min/w_s
    n_min_temp = max(n_min_temp,-1)
    n1.append(n_min_temp)

plt.plot(v,n, label = 'positive stall curve')
plt.plot(v,n1, label = 'negative stall curve')
plt.plot([v[0], v[0]], [n[0], n1[0]], color='black', alpha=0.5)
plt.plot([v[-1], v[-1]], [n[-1], n1[-1]], color='green', alpha=0.5, label =
    'red-line speed')
plt.title("V-n Diagram")
plt.xlabel("V")
plt.ylabel("n")
plt.ylim(-3, 4)
plt.legend(loc = 'upper left')
plt.grid()
plt.show()
```

Appendix B

Previous Aircrafts

B.0.1 Believer

- MTOW = 5.5 Kg.
- Empty Weight = 2.4 Kg.
- Range = 15 Km.
- Endurance = 2 hr.
- Length = 1.07 m.
- Wingspan = 1.96 m.
- Wing Area = 0.43 m^2.
- Aspect Ratio = 12.39.
- Payload Weight = 0.67 kg.
- 2x 14,000 mAh 6S LiPo Battery.
- Cruise speed = 20 m/s.

B.0.2 Raven B

- MTOW = 2.2 Kg.
- Empty Weight = 1.82 Kg.
- Wingspan = 1.4 m.
- Wing Area = 0.286 m^2.
- Aspect Ratio = 6.85.
- Length = 0.9 m,
- Endurance = 75 mins.
- Cruise speed = 9 m/s.
- Altitude = around 70 m.
- Payload Weight = 0.38 kg.

B.0.3 Talon GT

- MTOW = 2 Kg.
- Empty Weight = 1.722 Kg.
- Wingspan = 1 m.
- Wing area = 0.14 m^2.
- Aspect Ratio = 7.14
- Wing Loading = $9.28 Kg/m^2$.
- Cruise speed = around 15 m/s.
- Length = 66 cm.
- Endurance = 45 mins.
- Li-ion 4S2P 35E 14.4V Battery.

B.0.4 PUMA LE

- MTOW = 10.7 Kg.
- Wingspan = 4.6 m.
- Wing Area = 2.048 m^2.
- Aspect Ratio = 10.33.
- Length = 2.1 m.
- Endurance = 6 hrs.
- Range = Up to 125 Km.

B.0.5 Bormatec Explorer

- MTOW = around 4Kg.
- Wingspan = 2.2 m.
- Length = 1.4 m.
- Endurance = 45 mins.

B.0.6 Albatross UAV

- MTOW = 10 Kg.
- Empty Weight = 4.4 Kg.
- Wingspan = 3 m.
- Wing Area = 0.68 m^2.
- Aspect Ratio = 13.6.
- Endurance = 4 hrs.
- Range =100 Km.
- Cruise Speed = 19 m/s.
- LiPo batteries 6S 8Ah.
- Length = 74 cm.

B.0.7 Bormatec MAJA

- MTOW = 3 Kg.
- Wingspan = 1.8m/2.2m.
- Length = 1.2 m.
- Endurance = 1 hr.

B.0.8 Sirius Pro

- MTOW = 2.7 Kg.
- Empty Weight = 1.7 Kg.
- Endurance = 50 min.
- Wingspan = 1.63 m.
- Wing Area = 0.324 m^2.
- Aspect Ratio = 8.2.
- Range = 55 Km.
- Cruise Speed = 18.05 m/s.
- Battery weighing about 700 grams.
- Length = 120cm.

B.0.9 Mini Shark UAV

- MTOW = 5.5 Kg.
- Empty Weight = 2.8 Kg.
- Endurance = 2 hrs.
- Wingspan = 2.6 m.
- Wing Area = 0.43 m^2.
- Aspect Ratio = 15.72.
- Cruise Speed = 15.28 m/s.
- 21000 mAh battery.
- Length = 1.25 m.

B.0.10 LockHeed Martin Desert Hawk III

- MTOW = 4 Kg.
- Empty Weight = 2 Kg.
- Endurance = 90 mins.
- Wingspan = 1.2 m.
- Wing area = 0.22 m^2
- Aspect Ratio = 6.54
- Maximum Cruise Speed = 25.55 m/s.
- Range = 15 Km.
- 1.5 Kg Battery.
- Length = 0.91 m.

B.0.11 AR3 Tekever

- MTOW = 22 Kg.
- Emoty Weight = 14 Kg.
- WingSpan =3.2 m.
- Endurance = 10 h.
- Cruise Speed = 33.33 m/s.

Appendix C

Contribution of each member

C.1 Week 1

- AE21B001 - Preliminary configuration, data collection
- AE23M019 - Propulsion system
- AE21B048 - Mission objective, mission profile
- AE23M043 - Difficulty faced with Hybrid Vtol
- AE23M028 - Some of the weights for aircrafts
- AE23M003 - Discussions regarding configuration

C.2 Week 2

- AE21B001 - Code for iterations, Energy calculation for all phases, Literature review and data collection.
- AE23M019 - Logic for code iterations.
- AE21B048 - Looked up features of camera
- AE23M043 - Theory for method of iteration.
- AE23M028 -
- AE23M003 - Looked up camera

C.3 Week 3

- AE21B001 - Code for $\frac{L}{D}_{max}$ vs $\sqrt{AR_{wet}}$, Energy and power calculation for all phases other than climb, battery selection, motor and propeller selection
- AE23M019 - Energy and power calculation for climb
- AE21B048 -
- AE23M043 - Derivation of $\frac{L}{D}_{max}$ relation
- AE23M028 - Theory for Flat plate skin friction coefficient
- AE23M003 -

C.4 Week 4

- AE21B001 - Wing loading for all phases apart from Climb
- AE23M019 - Wing loading for climb
- AE21B048 -
- AE23M043 -
- AE23M028 -
- AE23M003 -

C.5 Week 5

- AE21B001 - Code for the iterations
- AE23M019 - Theory for the process of iterations
- AE21B048 -
- AE23M043 -
- AE23M028 -
- AE23M003 -

C.6 Week 6

- AE21B001 - XFLR5 analysis of airfoils and finalizing one, XFLR5 analysis of wing, preliminary control surfaces sizing, Wing incidence calculation.
- AE23M019 - High Lift Devices and XFLR5 analysis on high lift devices.
- AE21B048 - Choosing Wing Type
- AE23M043 - Airfoil selection (6.2.1)
- AE23M028 - Wing Location
- AE23M003 - Theory for Wing Dihedral, sweep and taper ratio.

C.7 Week 7

- AE21B001 - Tail configuration, Data collection, Horizontal Tail,
- AE23M019 - Vertical Tail, iterations for Fuselage length, width and height
- AE21B048 - Component layout
- AE23M043 -
- AE23M028 -
- AE23M003 -

C.8 Week 8

- AE21B001 - Configuration of Landing Gear, Landing gear height, main landing gear position, ground clearance and takeoff angle, Static and Dynamic Loading
- AE23M019 - Struts
- AE21B048 - Tire sizing
- AE23M043 - Wheel Base,
- AE23M028 - Wheel Track
- AE23M003 - Preliminary Calculations

C.9 Week 9

- AE21B001 - Weights of components - theoretical weights of wing, Horizontal tail, fuselage, electronics, CAD for structure, estimation of CG, recalculation of everything from first weight estimation as there was a MTOW change, weight estimation of all components from CAD, report writing, formatting.
- AE23M019 - CAD model for landing gear and internal component placement.
- AE21B048 - Worked on layout i.e checked some design constraints
- AE23M043 - Theoretical weight estimation of landing gear and vertical tail
- AE23M028 -
- AE23M003 -

C.10 Week 10

- AE21B001 - Trim condition, Lateral Stability,
- AE23M019 - Control Surface sizing
- AE21B048 - Longitudinal Stability
- AE23M043 -
- AE23M028 -
- AE23M003 - Aileron theory

C.11 Week 11

- AE21B001 - C_{D_0} estimation, P vs V plot, V-n diagram, Range and Endurance, final formatting.
- AE23M019 -
- AE21B048 - Power Requirement, Climb Rate and Climb Angle, final formatting.
- AE23M043 -
- AE23M028 -
- AE23M003 -

C.12 Acknowledgement

1. AE21B001 -
2. AE23M019 -
3. AE21B048 -
4. AE23M043 -
5. AE23M003 -
6. AE23M028 -

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