AS5213 - Design of UAV/MAV

Commercial Delivery with VTOL

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Chapter 1

Mission Definition and Requirements

There has been a rise in demand for delivery-based goods in different regions across the country. This has caused an increase in traffic congestion in urban regions, which led to an increase in the total delivery time and the risks of potential road accidents due to high speeding as employees are constrained for scheduled delivery of goods. Moreover, the rural and remote regions of the country are deprived of delivery services due to extremely congested housing conditions and underdeveloped road networks.

Apart from these primary concerns some of the secondary concerns revolve around optimizing delivery operations to reduce environmental impact, offer flexible routing options, enhance safety, and potentially bring down human intervention. Therefore road-based delivery systems tend to not satisfy the needs of the remote areas of the country and will grow to be an obsolete technology in urban regions due to their inefficiency which include unscheduled services and environmental concerns.

As per the Global Status Report on Road Safety 2015[27], 1.25 million road traffic deaths occur every year. With just nothing but a bike and a smartphone, many from the younger generation are now attracted to the incentive-laced job of food delivery. In a rat race to provide food to customers faster than their competitors, delivery boys are frequently seen driving rashly, leading to increased risks of accidents. Apart from this, the statistics for increasing carbon footprint can be justified a survey in 2021, according to which the typical home delivery round for a courier covering fifty miles [28], with an assumption of 120 drop-offs and an average of 0.42 miles between each drop-off point, there is a carbon footprint of around 181g of CO_2 per parcel [29]. This number will further tend to increase to meet the growing demands of delivery services.

1.1 Mission Profile

1.1.1 Mission Statement:

The primary objective is to **design a dual propulsion electric fixed-wing VTOL for the delivery of commercial goods in urban and rural areas with the implication of revolutionizing the current logistic landscape.**

1.1.2 Scope of our proposed solution:

The initial scope of our proposed UAV delivery solution will target deliveries within a designated urban area, focusing on densely populated neighborhoods and commercial districts. We believe this initial focus allows for concentrated efforts in optimizing delivery routes and regular traversal within a manageable area. As we acquire data and experience, the scope can gradually be expanded to encompass wider territories, including peri-urban and potentially rural areas.

As we gather more data and gain experience, the scope can be further expanded geographically, functionally, and in terms of payload capacity. Future iterations may explore integration with existing delivery networks, delve into long-distance deliveries, and adapt to diverse weather conditions and terrains.

1.1.3 Configuration

The configuration chosen is the **Dual propulsion electric fixed-wing VTOL**, such that it gives the ability of copter like to hover and vertical takeoff and get the advantage of the fixed wing plane necessary in our mission. The quad plane-like vertical configuration is chosen because of the fewer moving parts than the other counterparts such as tilt-rotor, tail-sitter, etc.

Figure 1.1: Some VTOL in our config

1.1.4 Mission Requirements:

The mission is to build a Dual propulsion Electric Fixed-wing VTOL for the delivery services of goods across a wider range across the country.

1. **Payload Capacity and its safe delivery within scheduled time.** Commercial goods delivery can include a wide variety of items from medical supplies, food, and groceries to other breakable or sensitive equipment. Thus the VTOL should have a good payload capacity to deliver a large variety of goods. Moreover, the payload should not be affected by different stages of flight and unstable wind conditions.

2. **Ability to maneuver in confined spaces.** As the VTOL aims to give delivery access to all regions including where there is a possibility of confined housing conditions, it should be able to transit with minimal movement. Since the airstrip and take-off area can't be maintained in all terrains and cities, vertical take-off and landing is a suitable option.

3. **Easy Recharging/Fueling .** Since the VTOL's primary aim is the commercial delivery of goods in scheduled time, wider range, and less energy consumption, it should be easily rechargeable so that it's ready for the next delivery mission in a minimal

time. Furthermore, it should be able to perform multiple deliveries according to its payload capacity without any break.

4. **Maintain level flight in all weather conditions.** Since the delivery services should not be bound by bad weather conditions, they should maintain level flight and stability to deliver the payload safely.

5. **Payload drop Mechanism.** We will be developing a mechanism for payload delivery while the VTOL is hovering. This reduces the chances of payload getting damaged as it might be in cases where landing is necessary. The mechanism will be such that it ensures a soft drop of the package so there is no possibility of damage.

6. **Easily accessible Payload region.** It is required that the payload region is easily accessible to quickly and easily move the payload in and out without much difficulty.

7. **Navigation and collision Avoidance.** Navigation is required for locating the delivery point and providing the goods at the desired location. To prevent collision due to obstructing buildings or a flock of birds, the VTOL should be able to cruise with a level flight in different altitudes along with a proper collision avoidance system, therefore to prevent any damage and failure.

8. **Scalability and fleet Management.** Since at a larger scale of delivery VTOLs, it would be required to monitor and control multiple VTOLs, thus we need to implement a fleet management system for the same.

1.1.5 Mission Segment Details

The following are the stages for the Quadplane Electric Fixed VTOL UAV.

1. **Vertical Takeoff:** In this stage the VTOL takes off with the payload vertically with the help of rotors.

2. **Transition:** After reaching the cruising altitude and initial velocity the rotors are stopped and the VTOL transitions its motion to horizontal motion. The cruise speed is reached at this stage.

3. **Cruise (Forward):** This is the stage where VTOL covers most of its distance and navigates to the path for good delivery, maintaining its level of flight and stability.

4. **Transition:** After reaching the delivery location the VTOL lowers its altitude and so transitions to a vertical downward motion with the help of rotors.

5. **Hover and Delivery:** When the VTOL is at a low enough altitude it hovers until the good is safely delivered to the customer.

6. **Transition:** After the delivery is done, the VTOL comes back to its initial safe altitude where the cruising speed is reached and it transits to horizontal motion.

7. **Cruise (Return):** The VTOL cruises the rest of the distance before reaching the landing location.

8. **Transition:** After the landing point is reached, the horizontal velocity is brought to zero and vertical thrusters are turned on to balance the weight.

9. **Vertical Landing:** This is the last stage where the VTOL lands vertically with the help of vertical rotors after completing its delivery mission.

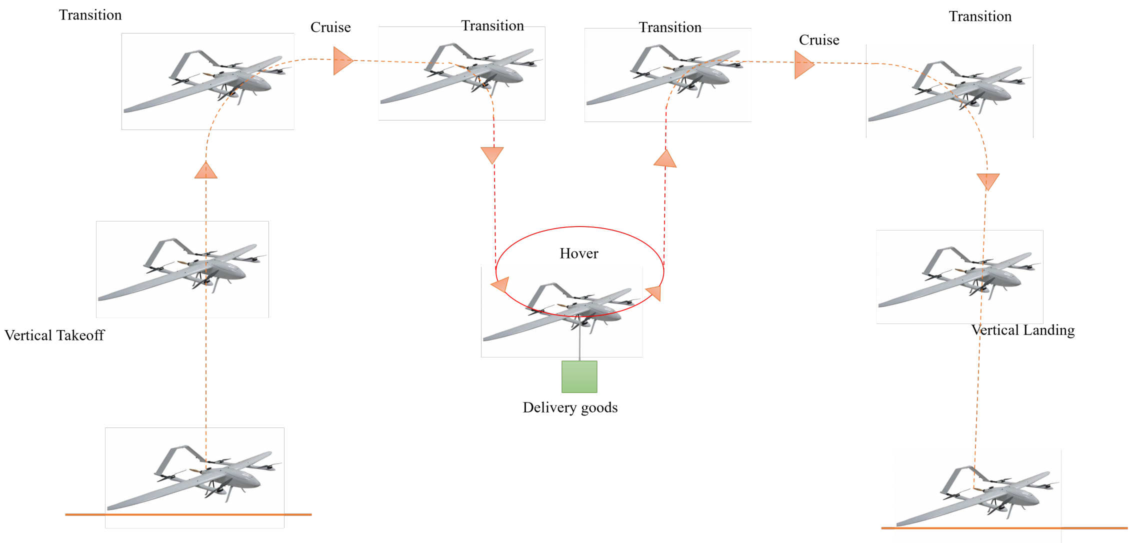


Figure 1.2: Mission profile

Segment	Time	Distance
Takeoff	70 sec	100 m (up)
Transition	30 sec	-
Cruise (Forward)	21 min	25 km
Transition	30 sec	-
Hover	70 sec	-
Transition	30 sec	-
Cruise (Return)	21 min	25 km
Transition	30 sec	-
Landing	40 sec	100 m (down)

1.2 Our Desired values

From the datasets collected we expect the following design values. The payload is so chosen such that most of the delivery goods are less than 5kg. Range is chosen such that we could deliver many items on one full charge. And the value of cruise is taken from the datasets.

Design Parameter	Value
Payload	3kg - 5kg
Range	150 km - 250 km
Maximum Endurance	1.5 hrs - 2.5hrs
Cruise Velocity	75kph - 90kph

Table 1.1: Desired values

1.3 Our Estimated Payload

From the common examples of deliveries made by commercial UAVs we have come up with an estimate for the maximum payload capacity for our fixed wing VTOL. Most of the deliveries made will not be in the range of our maximum payload, but this is kept as a safety and an emergency factor in case of a larger delivery requirement.

Commercial Products	**Average weight.**
Mobile phones or Cameras	200 g -1 kg
Books and other stationaries	500 g - 1.5 kg
Groceries	1 kg -2 kg
Clothing	0.2 kg - 0.4 kg
Food parcels	0.5 kg - 1 kg

Table 1.2: Common deliveries and their weights

1.4 Dataset

The dataset which includes information about the maximum payload, range, endurance, Maximum take-off weight, and so on from the pre-existing UAV is attached on the next page. Those pre-existing UAVs are of the same configurations as ours (i.e. Dual propulsion Electric Fixed Wing VTOL).
The $-$ in the dataset denotes the non-availability of the data. The Empty weight is found to be around 60 percent of the total weight for most of the UAVs.

Table 1.3: Datasets of existing UAV (Dual propulsion electric fixed-wing VTOL

Manufacture/Name	Max Payload	MTOW	Battery weight	Emp weight calc	Empty weight	Endurance	Range	Wingspan	Length	Airspeed(Cruise)	Airspeed(max)	Max ceiling
elevonX / Skyeye Sierra VTOL [1]	3 kg	12.5 kg	–	–	–	3 hrs	320 km	310 cm	152 cm	65 kmph	110 kmph	15km
elevonX / Tango VTOL [2]	5 kg	19 kg	–	–	–	2.5 hrs	420 km	300 cm	190 cm	72kmph	125kmph	15km
Penguin C MK2 [3]	4.5 kg	32 kg	–	–	–	12 hrs	180 km	410 cm	–	55kmph	110kmph	4km
Tekever AR3 VTOL [4]	4kg	25kg	–	–	–	16hrs	100km	350cm	150cm	85kmph	–	3.6km
FlyDragon/ FDG33 VTOL [5]	3 kg	18 kg	5.12	9.88 kg	9.7kg	3hrs	–	340cm	150cm	80kmph	110kmph	4km
FlyDragon/ FDG36 VTOL [6]	8 kg	30 kg	12.91 kg	9.09 kg	–	3hrs	–	360cm	175cm	90kmph	–	4km
FlyDragon/ FLY350 VTOL [7]	3 kg	15 kg	3.36 kg	8.64	9kg	2hrs(3kg)	–	350cm	160cm	72kmph	110kmph	4.5km
FlyDragon/ FDG410 VTOL [8]	10 kg	30 kg	7.2 kg	12.8 kg	16kg	2.3hrs(7kg)	–	350cm	190cm	65kmph	110kmph	4.5km
FlyDragon/ FDVF15 VTOL [9]	3.5 kg	15 kg	6.46 kg	5.04 kg	–	2.5hrs(1.5kg)	–	260cm	140cm	90kmph	–	4km
FlyDragon/ Baby shark 220 VTOL [10]	3.5 kg	12 kg	5.14 kg	3.36 kg	–	2.5hrs	–	250cm	–	–	100kmph	–
Yangda / FW-250 [11]	6.5kg(with bat)	13 kg	4.5 kg	6.5	5.5kg	2hr(6.5kg)	–	250cm	140cm	–	100kmph	3.5km
Yangda / Skywhale [12]	10 kg	34 kg	7.2 kg	16.8 kg	16kg	2hrs(6kg)	–	400cm	190cm	80kmph	120kmph	4.5km
Yangda / FW-320 [13]	6 kg	23 kg	7.2 kg	9.8 kg	9.76kg	2.5hrs(6kg)	–	320cm	120cm	80kmph	100kmph	3.5km
Blue shark F320 [14]	5kg	24kg	–	–	6.5kg	1.5hrs(5kg)	–	320cm	126cm	90kmph	–	4.5km
Foxtech Great Shark 330 [15]	4 kg	23 kg	2.1 kg	16.9 kg	7.4kg	2hrs(3kg)	–	320cm	125cm	80kmph	100kmph	3km
Foxtech AYK 350 [16]	10 kg	35 kg	10.31 kg	14.69 kg	15.5kg	2.5hrs(7kg)	–	350cm	188cm	94kmph	–	3.5km
Foxtech AYK 250 [17]	2.7 kg	15 kg	5.15 kg	7.15 kg	9kg	3hrs(1.5kg)	–	250cm	126cm	93kmph	–	–
Foxtech Whale 360 [18]	5kg	30kg	–	–	13kg	4hr(1kg)	–	350cm	182cm	94kmph	–	–
Unmanned RC Azure [19]	8kg	35kg	–	–	–	8hrs(2kg)	–	320cm	175cm	90kmph	–	5.4km
Unmanned RC Swans [20]	3kg	13kg	–	–	–	3hrs(2kg)	–	392cm	226cm	90kmph	–	5km
Foxteck AYK 250 pro [21]	2.7kg	15.5kg	–	–	7.7kg	3.5hrs(1.2kg)	–	250cm	126cm	90kmph	–	2km
Foxteck cetus 240 [22]	1.6 kg		4	4.4 kg	5.1kg	3.5hrs	–	240cm	151cm	–	–	3km
Foxtech Saber 210 [23]	1.5kg	13kg	–	–	kg	–	–	215cm	100cm	–	–	3km

Figure 1.3: Sierra-SkyEye-VTOL-3 [1]

Figure 1.4: elevonX / Tango VTOL [2]

Figure 1.5: Penguin-C-Mk2-VTOL-UAS [3]

Figure 1.6: Tekever-AR3-UAV [4]

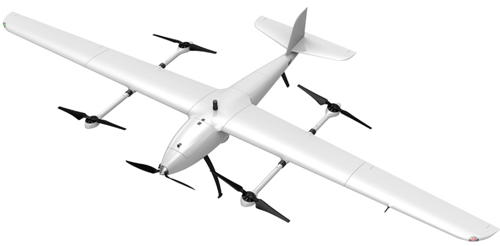


Figure 1.7: FlyDragon FDG33 VTOL [5]

Figure 1.8: FlyDragon FDG36 VTOL [6]

Figure 1.9: FlyDragon FLY350 VTOL [7]

Figure 1.10: FlyDragon FDG410 VTOL [8]

Figure 1.11: FlyDragon FDVF15 VTOL [9]

Figure 1.12: FlyDragon Baby shark 220 VTOL [10]

Figure 1.13: Yangda / FW-250 [11]

Figure 1.14: Yangda / Skywhale [12]

Figure 1.15: Yangda / FW-320 [13]

Figure 1.16: Blue-shark-F320 [14]

Figure 1.17: Foxtech Great Shark 330 [15]

Figure 1.18: Foxtech AYK 350 [16]

Figure 1.19: Foxtech AYK 250 [17]

Figure 1.20: Foxtech Whale 360 [18]

Figure 1.21: Unmanned RC Azure [19]

Figure 1.22: Unmanned RC Swans [20]

Figure 1.23: Foxteck AYK 250 pro [21]

Figure 1.24: Foxteck cetus 240 [22]

Figure 1.25: Foxtech Saber 210 [23]

Chapter 2

Preliminary Weight Estimation

Total Weight plays an important role in designing of a UAV. Power required, Wing design, and all other parameters depend on this value. So it is essential to have an initial estimate of the total weight of the UAV. For this calculation, the total weight(W_0) of the UAV is divided into three major weight categories. These are W_e, W_p and W_b

where,

W_e = Empty Weight of the UAV. This takes into account the weight of everything that doesn't change from mission to mission.

W_p = Payload Weight.

W_b = Battery weight.

$$W_0 = W_e + W_p + W_b$$
$$W_0 = W_0 \left(\frac{W_e}{W_0} + \frac{W_b}{W_0} \right) + W_p$$
$$W_p = W_0 \left(1 - \frac{W_e}{W_0} + \frac{W_b}{W_0} \right)$$

$$W_0 = \frac{W_p}{\left(1 - \frac{W_e}{W_0} + \frac{W_b}{W_0} \right)} \tag{2.1}$$

In equation 2.1, $\frac{W_e}{W_0}$ is called the Empty Weight fraction and $\frac{W_b}{W_0}$ is called the Battery weight fraction. We have to estimate both these fractions to determine a total weight.

2.1 Estimation of Empty Weight fraction

To estimate a value for the Empty weight fraction, we use the data from previously made UAVs for a similar mission profile. This dataset is compiled in Table **??**.
Based on historical trends, weights of similar aircraft follow the trend,

$$W_e/W_0 = A \cdot W_0^C \tag{2.2}$$

Now using the data collected, values of A and c can be found. Linear regression is performed on the data points of log(W_e/W_0) and log (W_0). The slope of it gives c and y-intercept gives A. This is performed using the code given in Appendix A.1 and the following plots are obtained.

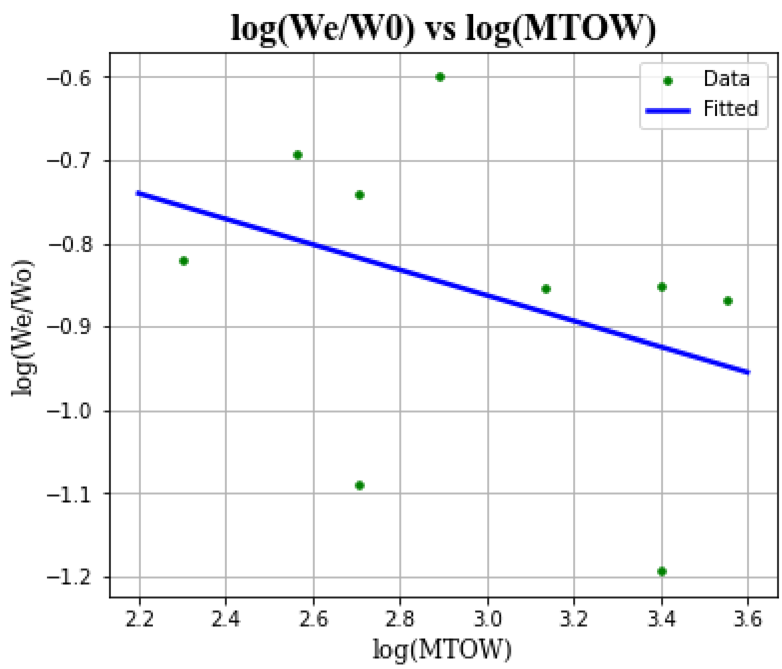


Figure 2.1: Linear Regression of $log(W_e/W_0)$ vs $log(W_0)$ plot

From fig2.1, we obtain A = 0.6684 and c = -0.1532.

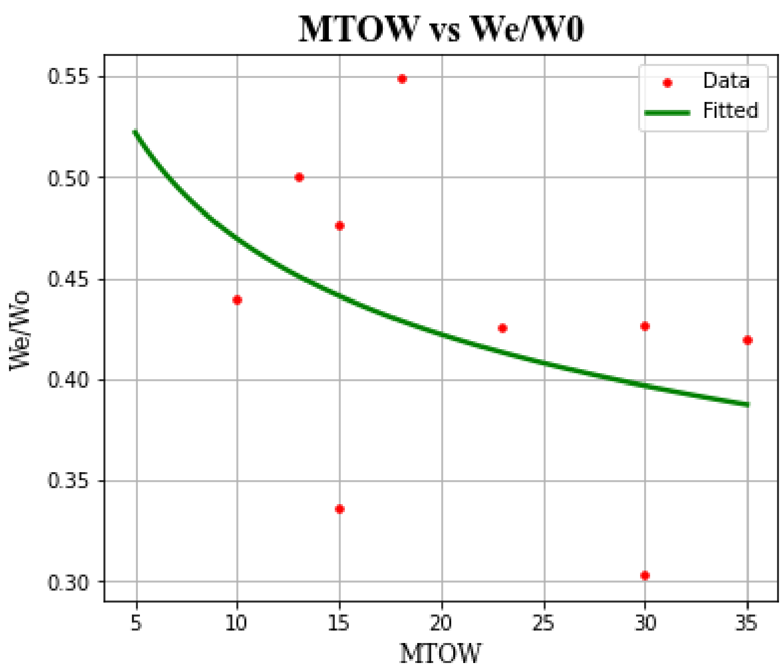


Figure 2.2: $\frac{W_e}{W_0}$ vs W_0 plot with obtained A and c values

Estimated Aerodynamic Parameters using previous dataset

2.2 Battery weight Estimation

The weight of the battery is calculated as a function of Wo, and power is estimated for each segment of the mission, which is then converted to energy using the estimated period for each segment.

2.2.1 Power required for Takeoff

The power required by rotorcraft can be derived based on momentum theory on the rotor disk. To realize vertical take-off and landing the ratio of thrust to weight(K) needs to be greater than one. The value of the K factor is taken as generally preferred 1.5. The maximum design power and thrust required of the VTOL-FW UAV can be written as

$$T_{TO} = W_{TO}K$$

Here the thrust is found for 1 rotor and the power is found for that rotor and the total power is multiplied by the number of rotors to get the total power [30].

$$P_{takeoff} = \frac{T_{TO}V_{TO}}{2}\sqrt{1 + \frac{2T_{TO}}{\rho V_{TO}^2 A_{Prop}}} \tag{2.3}$$

After substituting the values and take off weight obtained after the convergence plot we get:

$$P_{Takeoff} = 4.459 * 10^3$$

2.2.2 Power required for Transition

During the transition, we have assumed that while the vertical thrusters are on the rotor for horizontal cruise are powered to accelerate from zero to cruise velocity. Thus we can assume that the transition power is the average of hover power in addition to the power required to cruise from zero velocity to cruise velocity. The formula is taken from the research paper [30].

$$TransitionPower = \frac{HoverPower}{2} + \frac{\rho(V_{cruise}/2)^2 SC_d}{2}$$

$$TransitionPower = \frac{T^{3/2}}{FoM\sqrt{2\rho A}}/2 + \frac{\rho(V_{cruise}/2)^2 SC_d}{2} \tag{2.4}$$

After substituting the values and takeoff weight obtained after convergence plot we get:

$$P_{Transition} = 2.047 * 10^3$$

2.2.3 Power required for Cruise

The power required for a cruise can be found from the multiplication of cruise speed and thrust required. The thrust required is directly equal to the drag at cruise conditions.

$$P_{cruise} = V_{cruise}T_{cruise} \tag{2.5}$$

$$T_{cruise} = D = q_\infty SC_d \tag{2.6}$$

On substituting the estimated values we get:

$$P_{cruise} = 395.294W$$

2.2.4 Power required for Hovering

The power required for Hovering is derived from actuator disk theory, relative to the actual measured power required to produce thrust, where ideal power contains air density, , and rotor disk area, A. The measured power is the electric power supplied to the motor to drive the rotor and generate thrust. The formula used is given in article [30].

$$T_{hover} = W/4$$

$$P_{hover} = \frac{T^{3/2}}{FoM\sqrt{2\rho A}} \tag{2.7}$$

After substituting the values and take off weight obtained after convergence plot we get:

$$P_{hover} = 3.995 * 10^3$$

2.2.5 Power required for Landing

Landing required power must be lower than the power required for take-off. A descent velocity of = 1 m/s is taken, and then the propellers are working on a condition of low-speed axial descent < 2. Assuming the variable x = $-V_{Des}/V_h$, the actual induced velocity V_i at the propeller disk can be given by the following quartic approximation. The formula used is given in article [30].

$$V_{hover} = \sqrt{\frac{T_{hover}}{2\rho A_{prop}}} = \sqrt{89.9W_{TO}} \tag{2.8}$$

$$T_{hover} = WK \tag{2.9}$$

$$V_i = (K - 1.125x - 1.372x^2 - 1.718x^3 - 0.655x^4)V_H \tag{2.10}$$

$$P_{landing} = KW_{TO}(V_i - V_{des}) = 1.2W_{TO}(1.22(\sqrt{89.9W_{TO}} - 1) \tag{2.11}$$

After substituting the values and take off the weight obtained after the convergence plot we get:

$$P_{landing} = 1.872 * 10^3$$

2.3 Iterative Estimation of MTOW

2.3.1 Process

The coding part of the first weight estimation is done in python and added in the appendix. The process is as follows, First we take an initial guess of total weight. For that we find the empty weight fraction from the equation 2.2. We then get the W_b for that W_0 and that payload which is given in the Appendix A.5. With these values of W_0, W_b and W_p we use the equation 2.1 and code in Appendix A.4 to calculate the total weight of VTOL for these values.

We use the already found functions of Empty weight fraction in equation 2.2 and Battery weight estimation in section

Now we have a new value for W_0 which is estimated from the steps above. Lets now use this as the initial guess and carry forwards the steps mentioned above A.6, after a point of many iterations the W_0 converges (change in value is very less).

2.3.2 Parameters

The parameters used in the coding part of python for the VTOL is taken from the datasets added in the previous chapters as an estimate. The paramters used as mentioned below,

Table 2.1: Parameters used in Battery estimation

Paramters	Value
Wingspan	315 cm
Aspect ratio	10
No of rotors(Quadplane)	4
No of forward propellors	1
Density	1.2 kg/m^3
C_D (Taken as highervalue)	0.07
Diameter of propellor	18 in = 45.74 cm
Thrust to weight ratio(Takeoff)	1.5
Efficiency of propellor in cruise	0.85
Figure of merit (Hover)	0.7
No of runs	3.5

The number of runs is taken as 3.75 which was the optimal one. With the cruise velocity and time we get a range of 50 kms for one run. For 3.5 runs we get a range of around 175 kms The number of runs is not taken as a round number accounting for some efficiencies and safety factor.

Table 2.2: Mission segment details for battery estimation for 1 run

Parameters	Velocity	Time	Distance
Takeoff	1.5 m/s	70 s	100m (up)
Transition	0 - 20 m/s	30 s	300 m (forward)
Cruise	20 m/s	1250 s	25 km (forward)
Transition	0 - 20 m/s	30 s	300 m (forward)
Descent for hover	2.5 m/s	40 s	80 m (down)
Hover	-	70s	-
Ascent for cruise	1.5 m/s	50 s	80 m (up)
Transition	0 - 20 m/s	30 s	300 m (forward)
Cruise	20 m/s	1250 s	25 km (forward)
Transition	0 - 20 m/s	30 s	300 m (forward)
Landing	2.5 m/s	40 s	100 m (down)

2.3.3 Results

From the code A.1 and the parameters in the previous sections the result and converge of the total weight is given below,

From the graphs and Python code, The values of the aircraft are estimated as follows,

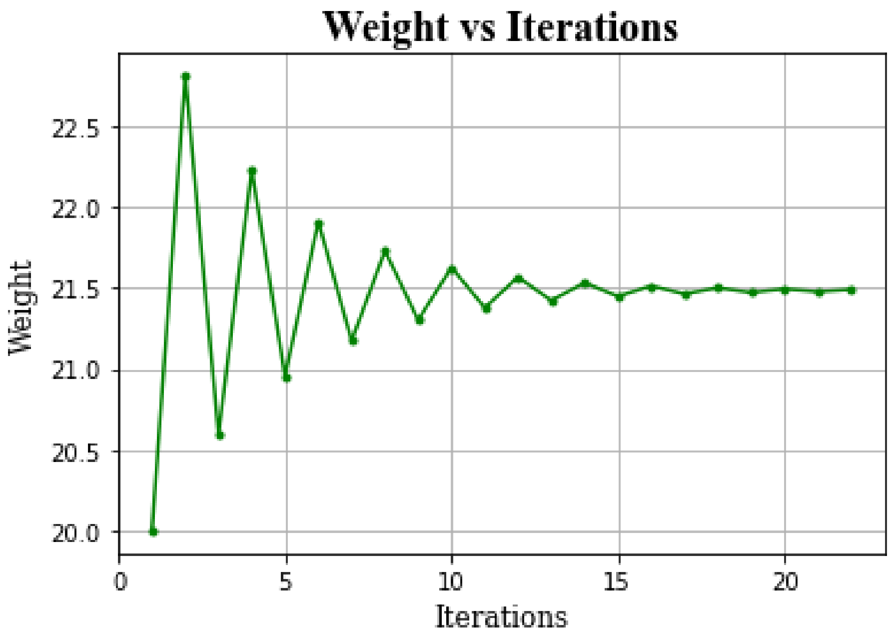


Figure 2.3: Total weight vs Iterations

The MTOW obtained from convergence	21.4 kg
The battery weight correspondingly	8.45 kg
The empty weight of the VTOL	8.95 kg
The payload weight of the VTOL	4.00 kg

The following table depicts the power and energy consumed in each segment from a Lithium-ion battery for the estimated time period.

Mission Segment	Power Required(W)	Time(sec)	Energy (Wh)
Take off	4459.77	70	86.71
Transition	2047.09	30	17.05
Cruise (Forward)	395.29	1250	137.254
Transition	2047.09	30	17.05
Descent to Deliver	1297.32	20	7.20
Hover	3995.37	70	77.68
Ascent	4459.77	35	43.35
Transition	2047.09	30	17.05
Cruise (Backward)	395.29	1250	137.254
Transition	2047.09	30	17.05
Landing	1297.32	40	14.41

Table 2.3: Table Depicting the Energy requirement in Wh for each segment

Thus total energy consumed in one run is **583.71Wh**

2.3.4 Battery Selection

For our Commercial Delivery VTOL, we are choosing Lithium-ion batteries due to their high energy density. Their balance between the ability for long-range flights and lightweight makes them ideal for our purpose Fixed Wing VTOL.

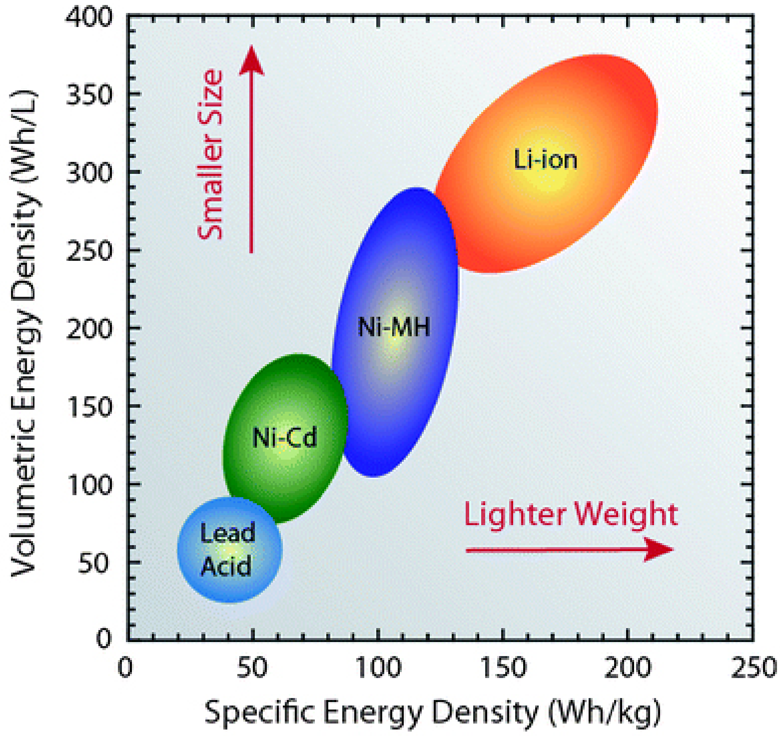


Figure 2.4: Energy cell density[24]

The energy density for Li-ion[24] can range from 150-270Wh/kg. Taking the energy density as 240Wh/kg according to the internet source, we get the following battery weight estimate in terms of Wo.

Thus according to our estimated energy and energy density of Lithium-ion, we get the following as our final battery estimate after convergence:

$$W_{Battery} = \frac{E_{1run} * N_{runs}}{Density} \tag{2.12}$$

$$W_{Battery} = \frac{(583.71) * (3.5)}{240} \tag{2.13}$$

$$W_{Battery} = 8.51kg \tag{2.14}$$

2.3.5 Some important points

The important points to be noted- in the processes are,

- **No of runs:** In the process of estimating the power and the battery weight we have considered more than a single run. This is to account for the calculation that involves multiple takeoffs and landings while delivering. Also since the takeoff and hovering take more energy, multiple runs are considered for a more accurate design.
- **In the process of delivering payload:** Sometimes the payload may not be delivered and so (or could be picked up for return after emptying the payload) the code has not been written by reducing the payload in the return cruise may be considered as an over-approximation too.
- **Takeoff/Hover/Landing:** While calculating the power and energy for the takeoff/landing/cruise the important part of the calculating is to divide the total thrust needed into each rotor and then find the power required for 1 rotor and multiply them by 4 to get the total power/energy needed in operation.

2.3.6 Safety factor

As a factor of safety, we add 10 percent to the total weight to get the following values as the final values.

The MTOW obtained from convergence	23.54 kg
The battery weight correspondingly	9.29 kg
The empty weight of the VTOL	9.845 kg
The payload weight of the VTOL	4.40 kg

Chapter 3

Power Loading and Power Plant Estimation

In this chapter, we will be estimating the Thrust loading, power loading, the motor to be used and the propellor to be used for the VTOL will be given.

The procedure that will be followed is as follows, first from the datasets (that was added in the first chapter) we will calculate all the other missing parameters, and then we will plot the graphs of $\frac{L}{D}$ vs $\sqrt{AR_w}$ which are calculated from the datasets. From the graph, we will pick a point estimate from our weight estimate and the weight of other vtols. From that estimated L/D, we recalculate the power and thrust in cruise and the other power estimated from the previous chapter.

From the estimated power and thrust for different segments, we will choose the motor and propeller and select the battery and number of cells as well. We will start this chapter by calculating the parasite drag.

3.1 Analysing previous dataset

By using the image processing methods, we have calculated the missing parameters such as MAC, taper ratio, fuselage area/diameter, and so on.

Table 3.1: Dataset recalculated

Name	MTOW(kg)	Cruise speed(m/s)	Wingspan(cm)	Wing area(cm^2)	root chord length(cm)	Tip chord length(cm)	AR
Foxtrot saber 210	13	20	215	7,141.73	41.9106	20.19	6.472521364
TEKEVER	25	23.6111	350	7,782.64	24.412	16.1287	15.74016015
FlyDragon/ FLY350 VTOL	15	20	350	7217.0237	28.793	14.6306	16.97375609
Cetus-240 VTOL	10	20	240	4,953.95	26.4205	16.4068	11.62708546
FlyDragon/ Baby shark 220 VTOL	12	22.2222	250	5758.3318	28.6893	16.6912	10.85383791
Yangda / FW-250	13	22.2222	250	5494.4724	26.545	17.6213	11.37506851
FlyDragon FDG410 VTOL	30	19.4444	412.8	13,923.45	45.8779	22.3109	12.23862189
Foxtech Great Shark 330	23	22.2222	320	8741.926	36.5249	18.7278	11.71366584
Foxtech AYK-250 Pro VTOL	15.5	25	250	5,919.04	28.9804	18.8792	10.55914473
Foxtech Whale 360 VTOL	30	26.1111	350	9,755.68	34.8119	20.5237	12.55678743
Foxtech AYK-350 VTOL	35	26.1111	350	11,618.31	39.2124	26.4231	10.54370214
Flydragon /FDG33 VTOL	18	22.222	340	7988.71	29.197	18.469	14.47042138

3.2 Wetted area calculation

Let's start the calculation of the wetted area of the aircraft data we have with some of the approximation formulas from Raymer and with the datasets we have.

3.2.1 Wing wetted area

For the wetted area calculation of the wing, the formula given below is used from Raymer [31] Chapter 7(eq 7.11) which is given below,

$$S_w = 2.003 S_{exp} \tag{3.1}$$

Here the formula used is such that the assumption of t/c <0.05 is made for the wing.

3.2.2 Fuselage wetted area

For the wetted area calculation of the fuselage, the formula given below is used from Raymer [31] Chapter 7(eq 7.13) which is given below,

$$S_w = \left(\frac{A_{top} + A_{side}}{2}\right) \tag{3.2}$$

By using these formulas we have estimated the wetted area for both the wings and the fuselage and have estimated the wetted area for each segment of the airplane.

Table 3.2: Wetted area calculated

Name	Wingspan(cm)	Wing area(cm^2)	Swet/Sref (wing)	Swet fuse	S/sref fuse
TEKEVER	350	7,782.64	2.003	11764	1.511569339
FlyDragon/ Baby shark 220 VTOL	250	5758.3318	2.003	3706	0.643589173
Yangda / FW-250	250	5494.4724	2.003	4080	0.742564473
FlyDragon FDG410 VTOL	412.8	13,923.45	2.003	10540	0.75699629
Foxtech Great Shark 330	320	8741.926	2.003	5327.8	0.609453798
Foxtech AYK-250 Pro VTOL	250	5,919.04	2.003	5660	0.956236146
Foxtech AYK-350 VTOL	350	11,618.31	2.003	11811.6	1.016636671
Flydragon /FDG33 VTOL	340	7988.71	2.003	6494	0.812897201

3.3 Friction coefficient

Since we have the wetted area for each component of the UAV we will have to determine the parasite drag for which the friction coefficient is needed. We will use the formula from Raymer [31] Chapter 12 (eq 12.27) which is used to estimate C_{fe} for a flat plate in turbulent conditions given below,

$$C_{fe} = \frac{0.455}{(\log R_e)^{2.58}(1 + 0.144M^2)^{0.65}} \tag{3.3}$$

from the previous chapters, we have our values of M = 0.05 and Reynolds number (R_e) changes according to the component as the fuselage has a different characteristic dimension than that of the wing.

3.4 Estimating the Zero lift drag

For the estimation of zero lift drag, we need the formula of the form factor to be used in the formula of the zero-lift drag coefficient, the formula is taken from Raymer [31] Chapter 12 (eq 12.24) is given below,

$$C_{Do} = \Sigma_i \left(C_{fe} F_i \frac{S_w}{S_{ref}}\right) + C_{Dp} \tag{3.4}$$

Where F is the form factor (accounts for pressure drag) for each component which will be discussed below. C_{Dp} is the drag due to the vertical propellers.

The form factor formula for the wing and fuselage is given below from Raymer [31] Chapter 12 (eq 12.30,12.31),

$$F_w = \left(1 + \frac{0.06}{(x/c)}\frac{t}{c} + 100(\frac{t}{c})^4\right)\left[1.34M^{0.18}(cos(\lambda)^{0.28})\right] \tag{3.5}$$

Which is estimated to be F_w = 1.1 with our approximations of thickness of the wing.

$$F_f = 1 + \frac{60}{(l_f/d_f)^{1.5}} + \frac{(l_f/d_f)}{400} \tag{3.6}$$

where L_f and D_f are the length and diameter of the fuselage respectively.

Using these formulas and our datasets the parasite drag is found to be as given in Table 3.4,

3.5 Estimating the drag polar

The drag polar estimated in this section is given below from Raymer [31] Chapter 12 (eq 12.47)

$$C_D = C_{Do} + KC_L^2 \tag{3.7}$$

Where C_{Do} involves C_{Dp} also and K is the constant that is given below,

$$K = \frac{1}{\pi(AR)e} \tag{3.8}$$

Here e is the Oswalds ratio that will be discussed below and AR is the aspect ratio of the wing.

The Oswals efficiency factor for a wing is approximated with the formula from the Raymer [31] Chapter 12 (Eq 12.48) since the sweep angle is low, we approximate the wings with the straight wing formula,

$$e = 1.78\left[1 - 0.045(AR)^{0.68}\right] - 0.64 \tag{3.9}$$

Table 3.3: Zero lift drag estimation

Name	fuselage diameter (cm)	length of fuselage(cm)	$\frac{l_f}{d_f}$	FF	MAC	Re wing	Cfe wing	Re fuse	Cfe fuse	Swet/S wing	Swet fuse	S/sref fuse	CD0
TEKEVER	30.163	170	5.636	1.2877775	0.2055	327882	0.0056	2712087.09	0.0037	2.003	11764	1.511569339	0.01950049
FlyDragon/ Baby shark 220 VTOL	20.4866	87.9524	4.2932	1.4728196	0.2322	348633	0.0055	1320606.61	0.0042	2.003	3706	0.643589173	0.01610713
Yangda / FW-250	22.2354	91.157	4.0996	1.5126039	0.2238	336091	0.0055	1368723.72	0.0042	2.003	4080	0.742564473	0.016910221
FlyDragon FDG410 VTOL	26.1322	192.6	7.3702	1.1683159	0.3545	465772	0.0052	2530405.41	0.0038	2.003	10540	0.75699629	0.01474667
Foxtech Great Shark 330	25.7801	110	4.2669	1.4779607	0.2858	429156	0.0053	1651651.65	0.0041	2.003	5327.8	0.609453798	0.015261592
Foxtech AYK-250 Pro VTOL	21.9339	126	5.7445	1.2775129	0.2429	410222	0.0053	2128378.38	0.0039	2.003	5660	0.956236146	0.016451913
Foxtech AYK-350 VTOL	29.4056	188	6.3933	1.2252822	0.3323	586320	0.0049	3316816.82	0.0036	2.003	11811.6	1.016636671	0.015389494
Flydragon /FDG33 VTOL	23.32	150	6.4322	1.2225774	0.2424	363892	0.0054	2252252.25	0.0039	2.003	6494	0.812897201	0.015812248

3.6 Estimating the drag due to vertical propeller

From the previous sections, we have calculated the drag polar, now to find the addition in drag due to the vertical propeller a diagram from research paper [25] is used to estimate the drag due to the vertical propellor.

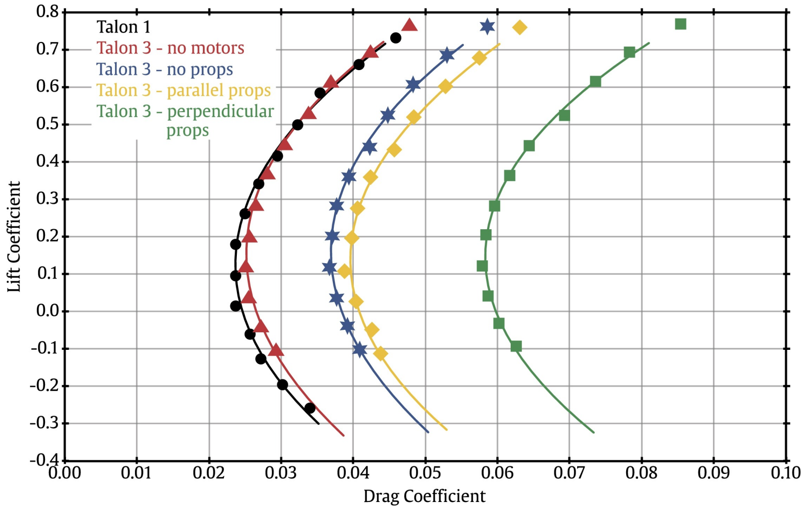


Figure 3.1: Drag due vertical propellor [25]

From the image inserted, we can see that there is an increment of 0.02 in C_D for a particular lift coefficient. So this is used to calculate the C_{Dp}

Table 3.4: Datasets with total drag coefficients

Name	MTOW(kg)	Cruise speed(m/s)	CD0	e	K	CL	CD	CD+CDp
foxtrot saber 210	13	20	0.0206	0.8547915	0.0576	0.744	0.0525	0.07248644
TEKEVER	25	23.6111	0.0195	0.6180863	0.0327	0.9421	0.0486	0.06855463
FlyDragon/ FLY350 VTOL	15	20	0.0188	0.590609	0.0318	0.8496	0.0418	0.06176417
Cetus-240 VTOL	10	20	0.0247	0.7152311	0.0383	0.8251	0.0508	0.07078625
FlyDragon/ Baby shark 220 VTOL	12	22.2222	0.0161	0.7346509	0.0399	0.69	0.0351	0.05512057
Yangda / FW-250	13	22.2222	0.0169	0.7215137	0.0388	0.7834	0.0407	0.06072201
FlyDragon FDG410 VTOL	30	19.4444	0.0147	0.7001641	0.0372	0.9318	0.047	0.0670123
Foxtech Great Shark 330	23	22.2222	0.0153	0.7130828	0.0381	0.8711	0.0442	0.06419267
Foxtech AYK-250 Pro VTOL	15.5	25	0.0165	0.7421677	0.0406	0.685	0.0355	0.05552302
Foxtech Whale 360 VTOL	30	26.1111	0.0112	0.6924208	0.0366	0.7374	0.0312	0.0511677
Foxtech AYK-350 VTOL	35	26.1111	0.0154	0.7425634	0.0407	0.7224	0.0366	0.05661842
Flydragon /FDG33 VTOL	18	22.222	0.0158	0.647099	0.034	0.746	0.0347	0.05473986

3.7 L/D vs Aspect ratio

From the datasets, we have all the values required to fit a curve between the L/D values and $\sqrt{AR_w}$, for which the values are tabulated and graphed as attached below,

Table 3.5: L/D vs $\sqrt{(AR)_w}$ values

Name	CL	CD_total	CL/CD	$\sqrt{(AR)_w}$
TEKEVER	0.942102104	0.068554633	13.74235499	2.803264639
FlyDragon/ Baby shark 220 VTOL	0.689965451	0.055120569	12.51738614	2.327829626
Yangda / FW-250	0.783357743	0.06072201	12.90072154	2.383068554
FlyDragon FDG410 VTOL	0.931753588	0.067012297	13.90421802	2.471870896
Foxtech Great Shark 330	0.871090936	0.064192665	13.56994503	2.418276417
Foxtech AYK-250 Pro VTOL	0.685043521	0.05552302	12.33800909	2.296010641
Foxtech AYK-350 VTOL	0.72242395	0.056618416	12.75952239	2.294331087
Flydragon /FDG33 VTOL	0.745999667	0.054739856	13.62808968	2.687819587

From this table we plot them to get the below images,

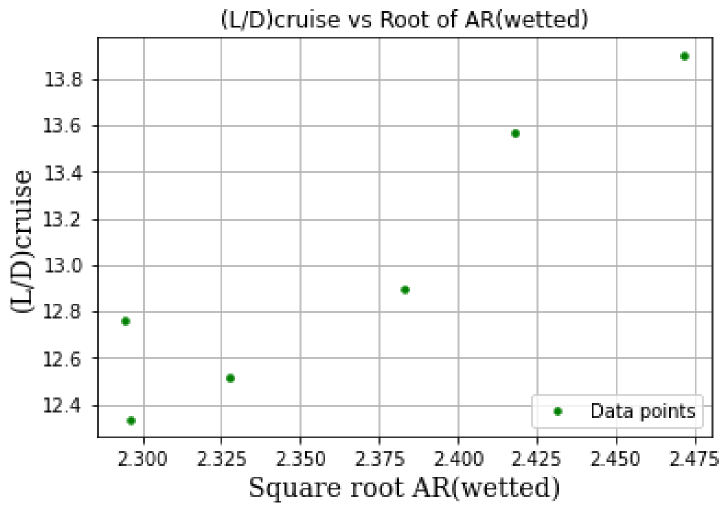


Figure 3.2: L/D vs $\sqrt{(AR)_w}$

Fitting the data points with a linear line we get the following graph, from that graph we take a suitable point which corresponds to the first estimated weight of our vtol.

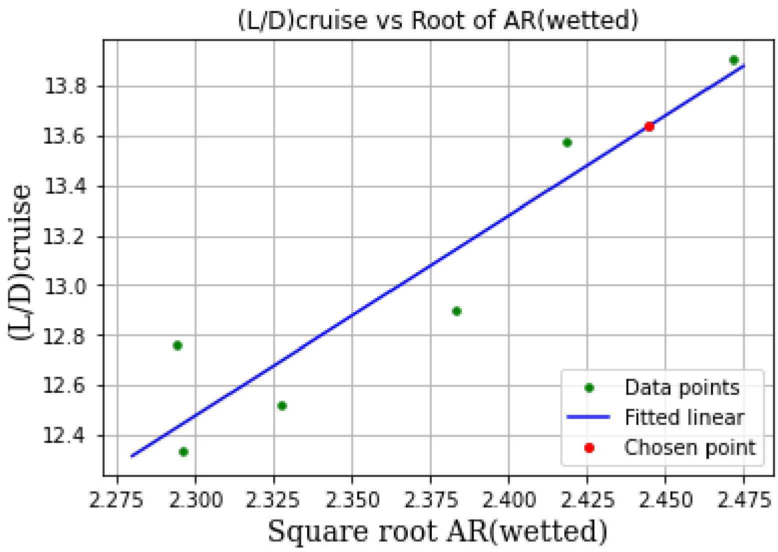


Figure 3.3: Fitted L/D vs $\sqrt{(AR)_w}$

From the graph and chosen point the corresponding values are,

The $\sqrt{(AR)_w}$:	2.445
The L/D ratio :	13.636
The lift in cruise :	23.54g

3.8 Thrust Estimation

Thrust estimation is done to get the required motor and propeller for our propulsion system. Since the ratio obtained for C_L/C_D is 13.636, the thrust-by-weight ratio at cruise according to Equation 3.1 is as follows:

$$(T/W)_{cruise} = 0.073 \tag{3.10}$$

Thus the **thrust in cruise is 16.88 N** according to the first weight estimate of 23.5Kg.

Thrust take-off required for a single rotor is calculated using the $(T/W)_{takeoff}$ =1.25 as calculated as 65.53N.

3.9 Second Iteration of Power Estimation

Here we perform the second iteration of power estimation since are drag coefficient is updated. As the obtained C_L/C_d is 13.636, the value of C_d is calculated as follows. For C_l

$$L = W = \frac{\rho v^2 S C_L}{2} \tag{3.11}$$

$$C_L = 0.959 \tag{3.12}$$

From the above ratio C_d **is obtained as 0.0703**.

With the updated value of the drag coefficient, the power required for the cruise and takeoff are updated as follows.

$$P_{cruise} = V_{cruise} T_{cruise} \tag{3.13}$$

$$T_{cruise} = D = q_\infty S C_d \tag{3.14}$$

On substituting C_d we get updated P_{cruise} as:

$$P_{cruise} = 396.988W$$

The Power takeoff is updated using the following equation:

$$P_{takeoff} = \frac{T_{TO} V_{TO}}{2} \sqrt{1 + \frac{2T_{TO}}{\rho V_{TO}^2 A_{Prop}}} \tag{3.15}$$

$$P_{takeoff} = 3.898 * 10^3 W \tag{3.16}$$

Thus power take-off required for a single rotor = 974.66W

3.10 Power Loading

Power loading refers to the ratio of the total weight of a UAV to its available power. The total weight of the UAV includes the payload weight, power available refers to the power generated by the UAV's propulsion system which in our case are electric motors.

3.10.1 Cruise Power Loading

Cruise Power loading is the power required per weight of the UAV to cruise at the desired altitude.

$$(W/P)_{cruise} = \frac{1}{T/W * (V/\eta)} \tag{3.17}$$

$$(W/P)_{cruise} = \frac{1}{0.073 * (20/0.85)} = 0.584 \tag{3.18}$$

3.10.2 Takeoff Power Loading

Take-off Power landing refers to the power required per weight of the UAV to take off to reach the desired altitude.

$$(W/P)_{take-off} = \frac{1}{T/W * (V/\eta)} \tag{3.19}$$

$$(W/P)_{Takeoff} = \frac{1}{1.25 * (1.5/0.85)} = 0.453 \tag{3.20}$$

Therefore $(W/P)_{Takeoff} < (W/P)_{Cruise}$

3.11 Powerplant Selection

Many commercially available motors and propellers were looked into and the best combination of these for our required power and thrust constraints has been chosen.

3.11.1 Motors

Vertical motors (for takeoff and hovering)

- Power calculated - 974.66 W
- Thrust calculated - 71.96 N

Based on the following constraints, the following motor (MN605S - KV170) 3.4[32] has been chosen

Figure 3.4: T-MOTOR MN605S KV170

The motor specifications are given in 3.5. Using a T-MOTOR G22*6.6 CF propeller given in 3.6, this motor gives the required 7.2kg at around 90% throttle. The power consumed is approximately 900W.

Type	Propeller	Throttle	Voltage (V)	Thrust (g)	Torque (N*m)	Current (A)	RPM	Power (W)	Efficiency (g/W)
MN605S KV170	T-MOTOR P22*6.6	40%	47.92	2067	0.52	4.30	3265	206	10.03
		42%	47.91	2208	0.56	4.70	3313	225	9.81
		44%	47.89	2384	0.60	5.20	3509	249	9.57
		46%	47.88	2553	0.65	5.70	3627	273	9.35
		48%	47.86	2704	0.69	6.30	3745	302	8.97
		50%	47.84	2892	0.73	6.90	3860	330	8.76
		52%	47.81	3067	0.77	7.50	3973	359	8.55
		54%	47.79	3273	0.81	8.20	4085	392	8.35
		56%	47.76	3471	0.85	9.00	4208	430	8.08
		58%	47.74	3584	0.88	9.60	4287	458	7.82
		60%	47.71	3736	0.94	10.20	4386	487	7.68
		62%	47.69	3896	0.99	10.90	4473	520	7.49
		64%	47.66	4104	1.03	11.70	4577	558	7.36
		66%	47.63	4283	1.08	12.60	4697	600	7.14
		68%	47.60	4500	1.12	13.50	4808	643	7.00
		70%	47.57	4721	1.18	14.30	4903	680	6.94
		75%	47.48	5477	1.30	17.40	5196	826	6.63
		80%	47.39	6064	1.44	20.20	5450	957	6.33
		90%	47.18	7201	1.72	26.20	5923	1236	5.83
		100%	46.97	8626	2.06	34.60	6469	1625	5.31

Figure 3.5: Motor Specifications

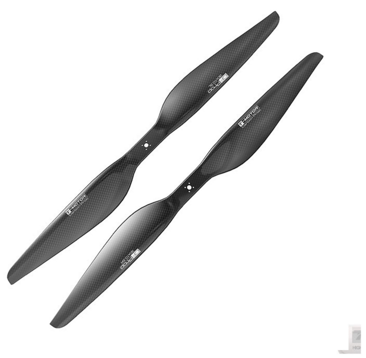


Figure 3.6: T-MOTOR G22*6.6 CF Prop

Cruise motor

- Power calculated - 396.988 W
- Thrust calculated - 16.899 N

Based on the following constraints, the following motor (TMOTOR MN601-S KV170) 3.7 [33] has been chosen

Figure 3.7: T-MOTOR MN601-S KV170

The motor specifications are given in 3.8. Using a T-MOTOR P20*6 CF propeller given in 3.9, this motor gives the required 1.721kg at around 50% throttle. The power consumed is approximately 144W.

Type	Propeller	Throttle	Voltage (V)	Thrust (g)	Torque (N*m)	Current (A)	RPM	Power (W)	Efficiency (g/W)
MN601S KV170	T-MOTOR P20*6	40%	47.96	1414	0.33	2.70	3272	129	10.92
		42%	47.95	1530	0.36	3.00	3389	144	10.64
		44%	47.94	1620	0.39	3.30	3506	158	10.24
		46%	47.93	1736	0.41	3.60	3624	173	10.06
		48%	47.92	1879	0.43	4.00	3753	192	9.80
		50%	47.90	1957	0.46	4.30	3851	206	9.50
		52%	47.89	2151	0.48	4.80	3994	230	9.36
		54%	47.88	2258	0.51	5.20	4121	249	9.07
		56%	47.85	2388	0.54	5.60	4225	268	8.91
		58%	47.84	2459	0.55	5.80	4228	277	8.86
		60%	47.82	2540	0.59	6.20	4362	296	8.57
		62%	47.81	2683	0.61	6.70	4481	320	8.38
		64%	47.79	2819	0.64	7.10	4589	339	8.31
		66%	47.76	2948	0.66	7.50	4690	358	8.23
		68%	47.74	3115	0.70	8.30	4791	396	7.86
		70%	47.73	3277	0.72	8.90	4923	425	7.71
		75%	47.67	3572	0.80	10.10	5144	481	7.42
		80%	47.62	3932	0.88	11.70	5395	557	7.06
		90%	47.48	4751	1.07	15.40	5901	731	6.50
		100%	47.34	5809	1.27	20.70	6482	980	5.93

Figure 3.8: Motor Specifications

Figure 3.9: T-MOTOR P20*6 CF Prop

3.12 Battery estimation

From this chapter and the previous chapter, the total energy in Watthour is around 2037Whr. From the motor and propellor configurations the Voltage that is required for the battery to produce is around 48V.

$$Q_{mah} = \frac{1000E_{Whr}}{V} \tag{3.21}$$

From which the mah of the battery is calculated to be 40740 mah. From the commercially available battery size we need One 45000 mah batteries for the optimal operation of the UAV.

The battery we require is of the configuration **12S 45000 Mah** Battery and the number of batteries is TWO. The photos of the battery are taken from the website.[34]

Figure 3.10: Battery specified

Chapter 4

Wing Loading Estimation

In this chapter, we will be estimating the wing loading to be used for calculations for the fixed wing VTOL. In simple terms, wing loading is the weight supported by a given area of the aircraft's wing. It is the Weight of the aircraft per unit wing reference area. Wing loading can be calculated as $\frac{W}{S}$. Here, W is the weight of the aircraft and S is the wing reference area.

While estimating the wing loading, we will find the suitable wing loading for all the segments and find an intersection like point such that it satisfies other segment constraints as well. The lower the wing loading is, the larger the wing will be the given weight and a larger wing can consequently increase the Drag and Weight.

For the estimation of wing loading, we consider the constraints on stall speed, maximum velocity, and the cruise condition, and turn as well.

4.1 Analysing previous data

From the previous dataset and our calculations, we get the following values,

Table 4.1: Datasets with Stall velocity and $C_{L_{max}}$

Manufacture/Name	MTOW	CD0	e	K	V(Cruise m/s)	Stall Velocity(m/s)	V_s/V_c	$C_{L_{max}}$
Tekever AR3 VTOL	25kg	0.0395	0.618	0.0327	16.66	–	–	–
FlyDragon/ FDG33 VTOL	18kg	0.035	0.647	0.034	22.22	–	–	–
Foxtech Great Shark 330	23kg	0.035	0.713	0.0381	22.22	18	0.814	1.3277
Foxteck cetus 240	10kg	0.0447	0.715	0.0383	19	15	0.7894	1.4688
Foxtech AYK 350	35kg	0.0354	0.742	0.0407	25	17	0.68	1.7043
Foxteck AYK 250 pro	15.5kg	0.0365	0.742	0.0406	25	15	0.6	1.9029

From this data we set the fraction V_{stall}/V_{cruise} = **0.75**. Also the $C_{L_{max}}$ = **1.6**. For V_{max} selection since, even though our motor is even capable of going past 30m/s, which could be seen from the fig.3.8 capable of 7kg thrust at 100% throttle, for which from the drag polar we calculated to be above 30 m/s. But from the research paper, we decided to keep V_{max} = **1.25** V_{cruise}

4.2 Parameters required

The parameters required are estimated from the previous datasets and from the conditions we set at each mission segment,

Parameter	Value
Zero lift drag coefficient(C_{D_0})	0.036
Lift-induced drag coefficient factor (K)	0.0349
Maximum lift coefficient ($C_{L_{max}}$)	1.6
Propeller efficiency (η_p)	0.85
Maximum velocity (v_{max})	25 m/s
Cruise velocity (v_{cruise})	20 m/s
Stall velocity (v_{stall})	15 m/s
Power Loading at cruise [$(W/P)_{cruise}$]	0.585
Thrust to Weight ratio at cruise [$(T/W)_{cruise}$]	0.073
Minimum Radius of turn [R_{min}]	50 m
Velocity at turn [V_{turn}]	22 m/s
Density of air (ρ)	1.225 kg/m^3

- **Zero lift drag coefficient(C_{D_0}) :** It is calculated by taking an average of C_{D_0} values of Tekever, Foxtech Great Shark 330, and Fludragon FDG33 VTOL respectively. These three UAVs are only considered because their MTOWs are very close to the MTOW considered for this UAV.(Note: C_{D_0} has C_p value included in it)

- **Lift induced drag coefficient factor (K) :** It is calculated by taking an average of aspect ratios of Tekever, Foxtech Great Shark 330, and Fludragon FDG33 VTOL respectively, Similarly, Oswald's efficiency factor is also estimated by taking the average values of 5 similar datasets. And substituting them in the following equation,

$$K = \frac{1}{\pi e A.R} \tag{4.1}$$

  Where, e = Oswald's efficiency factor
  A.R = Aspect ratio
  Hence, We estimated the K value as 0.0349

- **Minimum radius of turn(R_{min}):** Since we are delivering products, most of the time we will be flying in cities too. Hence, we will have to turn when there is a large building in the way, so we restrict the minimum turn radius to 50 m. So we will avoid an obstacle that is far by 50m when detected.

- **Velocity of turn(V_{turn}):** In a similar way to above, Our cruise velocity is 20 m/s and our maximum velocity is 25m/s. From flight dynamics, we enter a level turn by increasing the velocity keeping the $C_{L_{cr}}$ fixed. The increase in velocity increases the lift which compensates for the roll as the lift vector is tilted. Thus we choose a velocity 22 m/s that will have a roll angle of around 37 degrees.

4.3 Stall speed constraint

So here is how we will fix this constraint, We have the $C_{L_{max}}$ and V_s from the previous sections.

$$L = W = \frac{\rho C_{L_{max}} V_{stall}^2 S}{2} \tag{4.2}$$

So we can write,

$$\frac{W}{S} = \frac{\rho C_{L_{max}} V_{stall}^2}{2} \tag{4.3}$$

Considering C_{Lmax} as 1.6 and V_s as 15 m/s from previous data. We get the following wing loading.

$$\frac{W}{S} = 216N/m^2 \tag{4.4}$$

In this equation, we are fixing the stall velocity(V_s) and the C_{Lmax}. So from eq(4.3) when the wing loading is taken to be higher than 216 N/m^2 then as the C_{Lmax} is fixed we will be getting a higher stall velocity which is not good. So the wing loading found from this constraint sets the upper bound of the Wing loading selection.

4.4 Maximum forward speed constraint

As the speed of the aircraft increases, more lift is changed by a change in the angle of attack so a smaller wing (Higher Wing loading) is less adversely affected by vertical gusts, on the downside higher wing loading also reduces maneuverability. We have calculated the wing loading for a forward maximum speed of 25m/s.

$$W/P = \frac{\eta_p}{\frac{aV_{max}^3}{W/S} + \frac{b}{V_{max}(W/S)}} \tag{4.5}$$

where a and b are defined as follows:

$$a = \frac{\rho C_{Do}}{2} \tag{4.6}$$

$$b = 2k\rho_o \tag{4.7}$$

The power loading for Vmax was estimated as follows:

$$Q = \frac{\rho v_{max}^2}{2} \tag{4.8}$$

The C_{Dmax} evaluated at maximum speed is as follows:

$$C_{Dmax} = C_{Do} + K * C_{Lmax}^2 \tag{4.9}$$

$$T_{max}/W = Q * C_{Dmax}/W \tag{4.10}$$

$$W/P_{max} = \frac{\eta_p}{T_{max}/W * V_{max}} \tag{4.11}$$

The power loading at maximum forward speed is calculated as 0.4212. Substituting in equation (4.5) we get the following solutions for wing loading.

$$W/S = 0.45N/m^2 \& 917N/m^2 \tag{4.12}$$

4.5 Cruise condition constraint

At the cruise condition, we know the T/W ratio is 0.073 from Section 3.8. Here, T/W a cruise can be a function of wing loading ($\frac{W}{S}$) i.e,

$$\frac{T}{W} = \frac{\frac{1}{2}\rho v^2 C_{D_0}}{(W/S)} + \frac{2K}{\rho v^2}(\frac{W}{S}) \tag{4.13}$$

We know that the v = Cruise speed = 20 m/s, Density of air (ρ) = 1.225 kg/m^3 , C_{D_0} = 0.036, K = 0.0349.

Substituting these values in the equation (4.7), The wing loading solutions are

$$\frac{W}{S} = 194.13, 319.96 N/m^2 \tag{4.14}$$

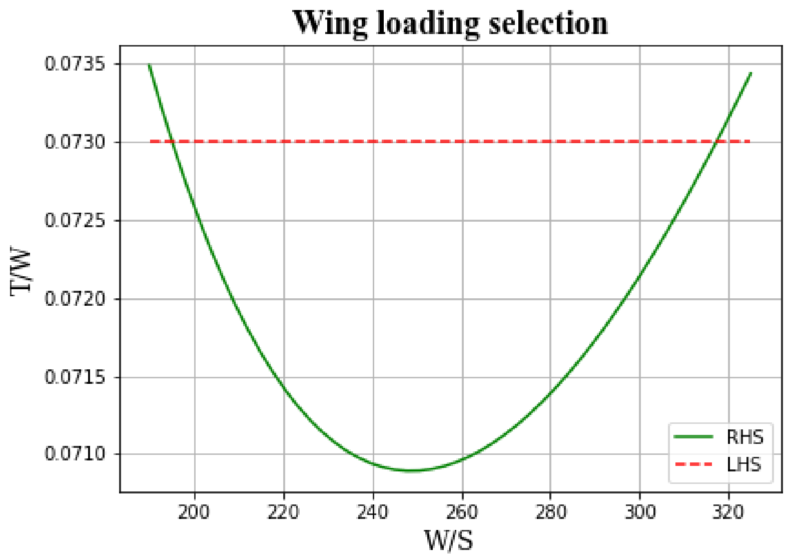


Figure 4.1: W/S Cruise condition

From the graph, The green line gives the left hand side of the equation which is similar to the non dimensional thrust required and the red dotted line gives the left hand side which is the fixed cruise non dimensional thrust. So we could see in the region between the intersections, we could see the thrust required is lesser than the thrust at cruise, so the middle region is the suitable wing loading.

4.6 Turn condition

Now the way we are going to set the turn condition is as follows, we fix a minimum radius of turn as 50m as seen before. we have also fixed the velocity of turn as 22 m/s. From J.D.Anderson [35] Chapter 6, eq(6.9) the equation for radius of turn is,

$$R = \frac{V^2}{g\sqrt{n^2 - 1}} \tag{4.15}$$

where n is the load factor(Lift to weight ratio).

From the definition of load factor, we have,

$$nW = \frac{1}{2}\rho V^2 S C_{L_{cr}} \tag{4.16}$$

$$\frac{W}{S} = \frac{\frac{1}{2}\rho V^2 C_{L_{cr}}}{n} \tag{4.17}$$

Substituting the load factor in eq(4.11) in eq(4.9),

$$R = \frac{V^2}{g\sqrt{\left(\frac{\frac{1}{2}\rho V^2 C_{L_{cr}}}{\frac{W}{S}}\right)^2 - 1}} \tag{4.18}$$

By substituting the expected values of the parameters such as V = 22 m/s, $C_{L_{cr}}$ = 0.9924 obtained in the last chapter, g = 9.81 m/s^2, R = 50m. (Note: Here we keep the lift coefficient same as that of the cruise because when we enter the horizontal turn, we do not change AOA we only increase the thrust).

We get the value of wing loading to be W/S = **205.13** N/m^2. To check whether this is upper bound or lower bound, we keep all parameters fixed we could see that the radius is directly proportional to the wing loading and so when the wing loading is decreased, the radius decreases which is not acceptable as we have fixed the minimum radius. So this value of wing loading sets the lower bound.

4.7 Final range of Wing loading

From the previous section, collecting all the values of the wing loading ranges we have the following table,

Table 4.2: Final conditions

Condition (Segments)	**Range of Wing loading(N/m^2)**
Stall condition	216 > W/S
Maximum velocity condition	0.45 < W/S < 917
Cruise condition	194.13 < W/S < 319.96
Turn condition	W/S > 205.13
Total condition	205.13 < W/S < 216

From this chapter the wing loading is found out to be,

$$205.13 < \frac{W}{S} < 216 \tag{4.19}$$

where the units are in N/m^2

Chapter 5

Second Weight Estimate

In this chapter, we will estimate the revised total takeoff weight of the VTOL, since we have some more data compared to the previous weight estimate, such as powerplant weight, estimated wing loading, etc. This time we will be able to estimate the airframe weight.

Here we will use some concepts from the previous chapter with revised values, such as the method of battery weight estimation method remains the same. The empty weight fraction from the total weight fitted curve in Chapter 2, fig.2.2 the method of carrying over iterations remains the same as Chapter 2, but with changed parameters and some methods which will be discussed in this chapter.

5.1 Parameters Required

The parameters required for the calculations of the following report are in the table below which are collected from datasets corresponding to powerplant selected from section 3.11. The weight of the motor and propeller combination is taken from the websites of the motor respectively.

Parameter	**Value**
Weight of Cruise Motor	800 gm
Weight of one Vertical Motor	650 gm
Weight of Electronic Speed Controller (ESC)	400 gm
Zero lift drag coefficient (C_{do})	0.036
Lift-induced drag coefficient factor (K)	0.0349
Drag Coefficient (C_D)	0.0616
Lift Coefficient (C_L)	0.8571
Ostwald's efficiency factor (e)	0.662
Loading factor (W/S)	210
Total Powerplant Weight	3800 gm

Total Powerplant Weight: It is the sum of weights of 4 vertical motors, 1 cruise motor, 1 horizontal propeller, 4 vertical propellers (weight of propellors is already added in motor weights) and Electronic Speed Controller.

$$\text{Total Weight} = 4*650 + 1*800 + 400 = 3800 \text{ gm}$$

Hence, the Total Weight of the Powerplant is 3800 gm (or) 3.80 kg.

5.2 Weight estimation

We already know that the total weight of the aircraft is the sum of the empty weight (the weight of the airframe), the weight of the payload the weight of the battery, and the weight of the motors and propellers.

$$W_o = W_{battery} + W_{payload} + W_{airframe} + W_{motor/propeller} \tag{5.1}$$

Unlike the first weight estimation, we need not use the battery weight fraction for the estimation of MTOW as the battery weight is known for the aircraft.

$$W_o - W_a = W_b + W_p + W_{m/p} \tag{5.2}$$

$$W_o(1 - \frac{W_a}{W_o}) = W_b + W_p + W_{m/p} \tag{5.3}$$

$$W_o = \frac{W_b + W_p + W_{m/p}}{1 - \frac{W_a}{W_o}} \tag{5.4}$$

In the above equation, W_o is the total weight or the MTOW. W_b is the battery weight, W_p is the payload weight, $W_{m/p}$ is the combined weight of the motors and propellers used in the UAV and W_a is the weight of the airframe.

To estimate a value for the airframe weight fraction $\frac{W_a}{W_o}$, we use the data from the previously collected data from other UAVs with a similar mission profile. We know that the empty weight is the sum of the weight of the airframe and the motors and propellers. So,

$$\frac{W_e}{W_o} = \frac{W_a}{W_o} + \frac{W_{m/p}}{W_o} \tag{5.5}$$

Based on historical trends, we can write,

$$\frac{W_e}{W_o} = A.W_o^C \tag{5.6}$$

In the above equation, A and C are found from the data we have. Like we performed in 2.1, Linear regression is done on the data points of $log(W_a/Wo)$ and $log(W_o)$. The slope of the plot gives C and the Y-intercept gives the A.

Once the value of $\frac{W_e}{W_o}$ is found, the subtract $\frac{W_{m/p}}{W_o}$ from it to get $\frac{W_a}{W_o}$.

5.3 Iterative Estimation of MTOW

The coding part of the second weight estimation is done in Python and added in the appendix B. The process is as follows, First, we take an initial guess of total weight. For that, we find the empty weight fraction from the equation 2.2. From that empty weight fraction and the powerplant weight fraction, we get the airframe weight fraction. We then have the W_b for that payload, powerplant estimation which is given in the Appendix B.4. With these values of W_0, W_b and W_p we use the equation 5.5 and code in Appendix C.2 to calculate the total weight of VTOL for these values.

Now we have a new value for W_0 which is estimated from the steps above. Let now use this as the initial guess and carry forward the steps mentioned above B.5, after a point of many iterations the W_0 converges (change in value is much less).

5.3.1 Results

The results plotted are attached as follows,

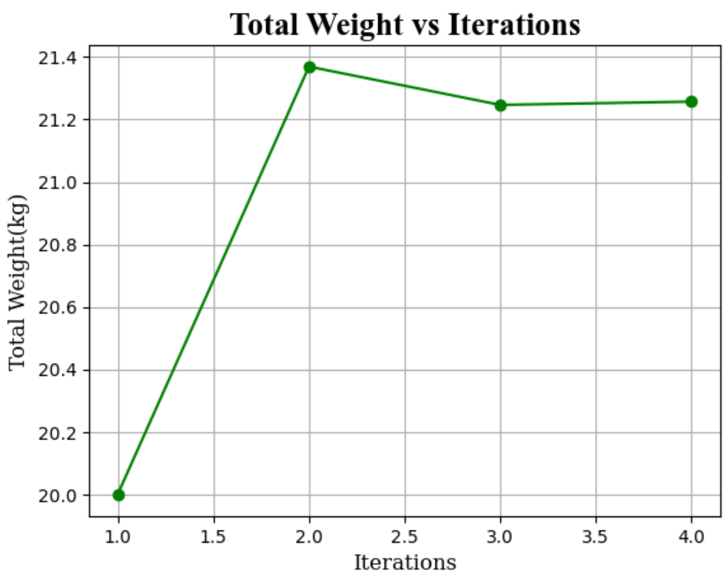


Figure 5.1: Total weight vs Iterations

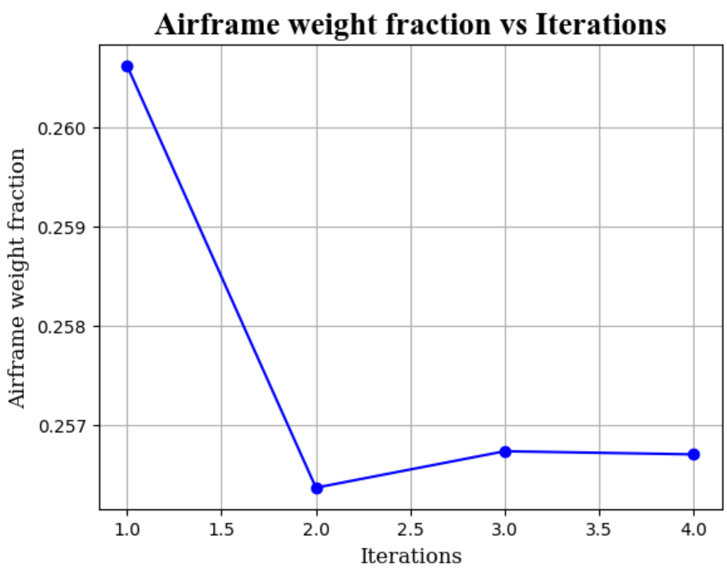


Figure 5.2: Airframe weight fraction vs Iterations

5.4 Second weight estimate results

From the graph and this chapter the weight estimated is as follows,

The MTOW obtained from convergence	21.256 kg
The battery weight correspondingly	8 kg
The powerplant weight of the VTOL	3.8 kg
The airframe weight of the VTOL	5.45 kg
The payload weight of the VTOL	4 kg

With a safety factor of 10 percentage we get the final values as,

The MTOW obtained from convergence	23.4 kg
The battery weight correspondingly	8.8 kg
The powerplant weight of the VTOL	4.15 kg
The airframe weight of the VTOL	6 kg
The payload weight of the VTOL	4.4 kg

Chapter 6

Wing Design

In this chapter, we will be selecting and estimating all the necessary parameters required for a wing. This involves the selection of airfoil, Aspect Ratio, Taper ratio, sweep, dihedral, geometric twist, and variation of thickness along the span. The airfoil section is responsible for generating the optimum pressure distribution in the top and bottom surfaces, essentially producing high lift and low drag. The other parameters are selected based on the dataset or references found.

6.1 Airfoil selection

The airfoil selected will be based on the required parameters such as $C_{l_{max}}$, $C_{l_{cruise}}$ and whether maximum C_l/C_d is attained at a C_l close to out operation range. Then, the lift and drag characteristics of many airfoils are looked upon to determine the best possible choice for us.

6.1.1 Estimation of required parameters

The required parameters found from the previous chapters are added as follows,

Parameter	**Value**
Weight	24.6 kg
W/S	210 kg/m^2
Cruise speed	20 m/s
$C_{l_{cruise}}$	0.8571
$C_{l_{max}}$	1.55
$(L/D)_{max}$	13.86

Table 6.1: Required parameters

The given parameters are for the entire wing. While the wing is a 3D body, the airfoil is a 2D section. Therefore, the lift coefficient values are modified using the following approximation for the airfoil:

$$C_{l_{cruise}} = \frac{C_{L_{cruise}}}{0.95}$$

$$C_{l_{cruise}} = 0.902$$

Similarly

$$C_{l_{max}} = 1.63$$

6.1.2 Various airfoils data

The airfoils (some of them) that were looked for when choosing the optimal one are attached below.

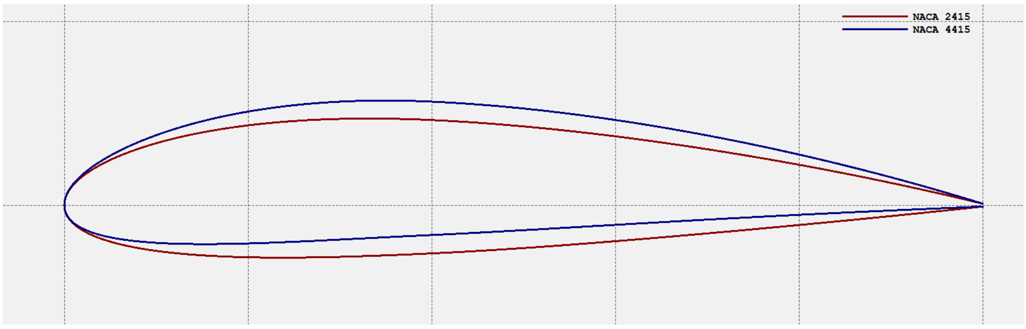


Figure 6.1: Airfoils plotted

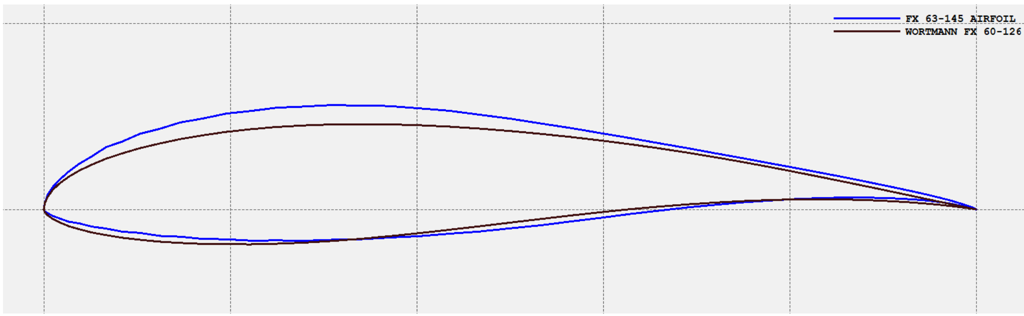


Figure 6.2: Airfoils plotted

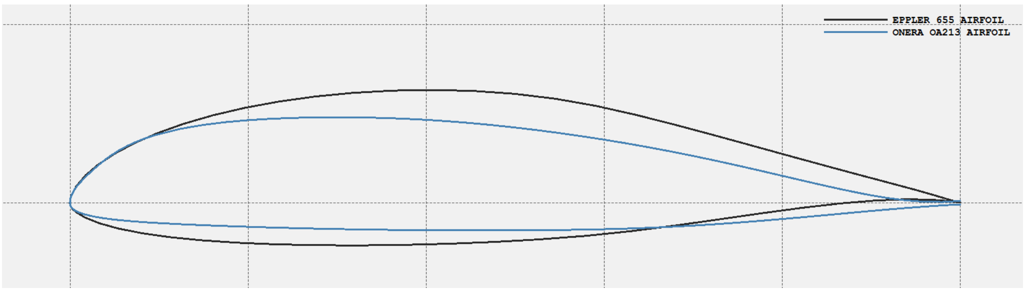


Figure 6.3: Airfoils plotted

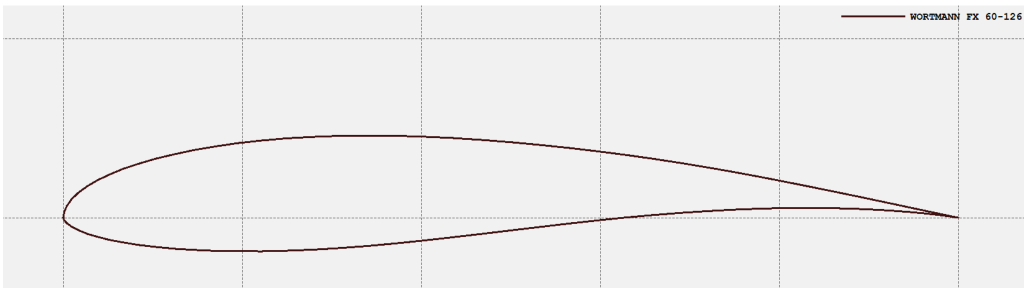


Figure 6.4: Chosen airfoils plotted (WORTMANN FX 60-126)

The plots of the different airfoils are attached (Simulations are done XFLR5 software) below,

Table 6.2: Co-efficient data of different airfoils at Re=400000 (typically @ MAC)

Name of the Airfoil	Cl_max	(CL/CD)_max	Stall angle (degrees)	CL/CD @ operational CL
NACA 2415	1.36	82.36	14.5	79.8
NACA 4415	1.52	95.2	13.1	85.05
ONERA OA213	1.51	70.2	13.5	68.1
EPPLER 655	1.35	104.5	11	60
Wortmann FX 63-145	1.5	70	12	57.8
Wortmann FX 60-126	1.63	97.25	14	87.45

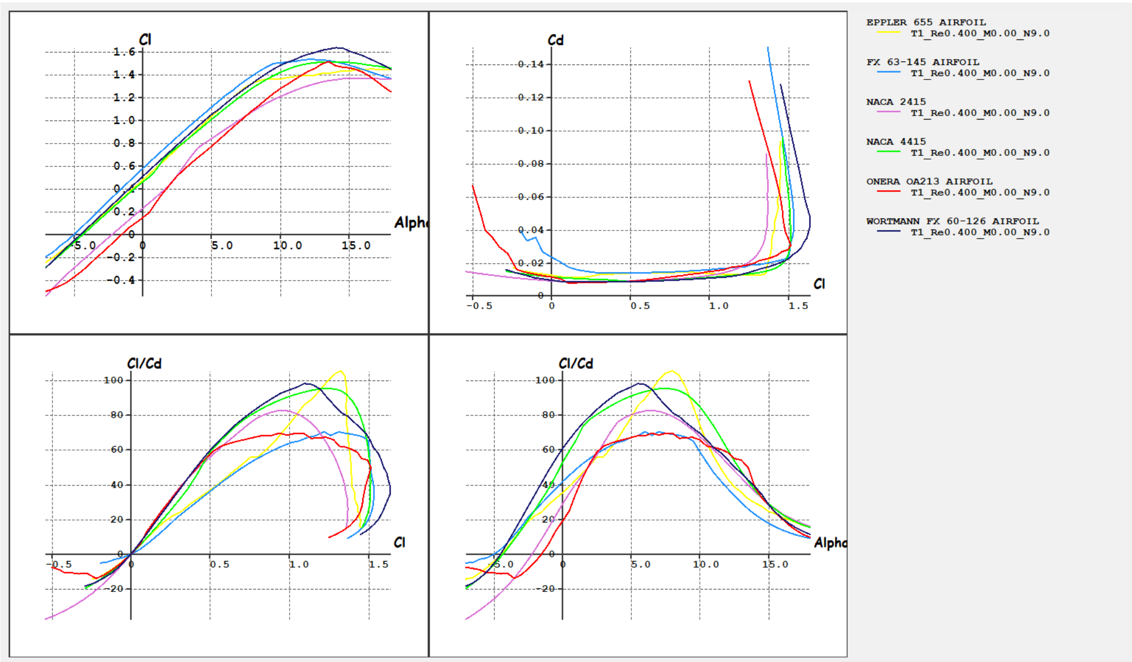


Figure 6.5: Plots of various airfoil

Considering the factors mentioned above, we chose the airfoil **WORTMANN FX 60-126** as our airfoil.

6.1.3 WORTMANN FX 60-126 Airfoil

The airfoil plotted separately is shown here,

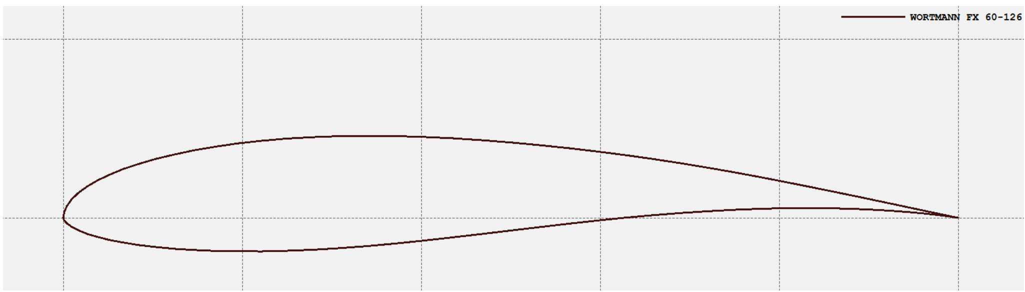


Figure 6.6: Chosen airfoils plotted (WORTMANN FX 60-126)

In this section, we will see the aerodynamic coefficients plotted for our chosen airfoil Co-efficient data of different airfoils at Re=400000 (typically @ MAC).

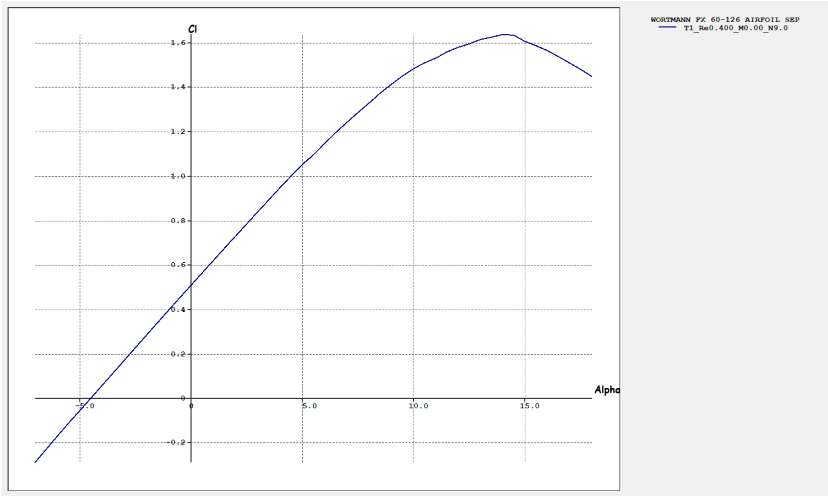


Figure 6.7: C_L vs α

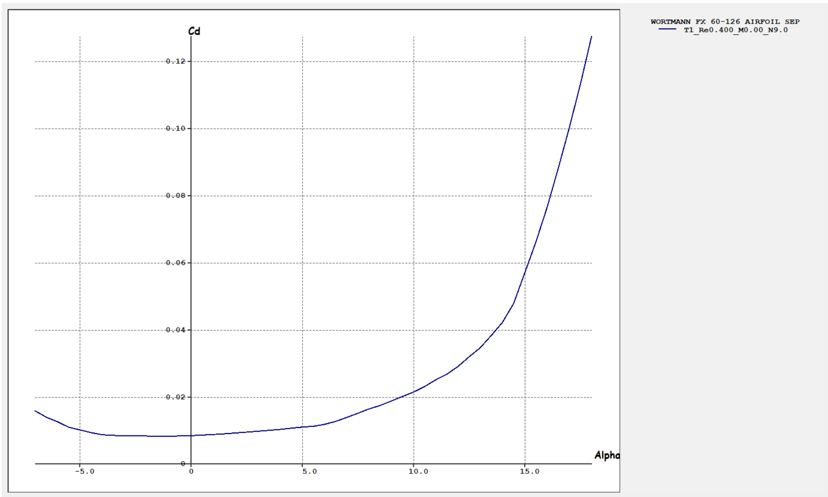


Figure 6.8: C_D vs α

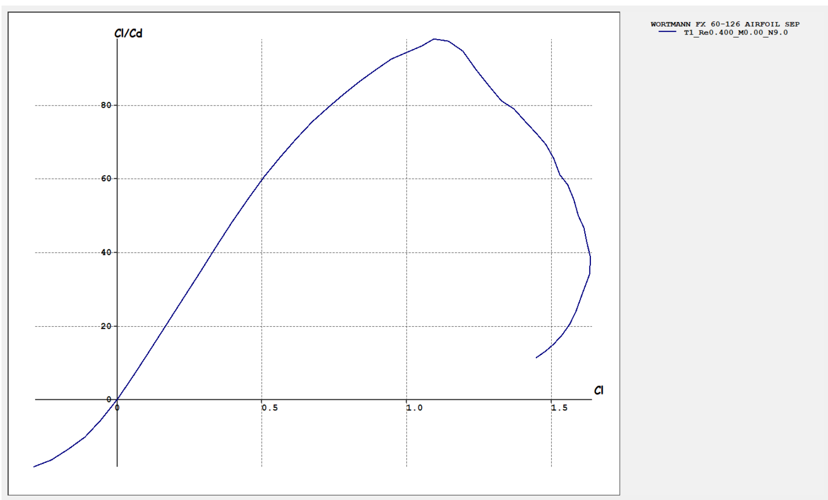


Figure 6.9: C_L/C_D vs C_L

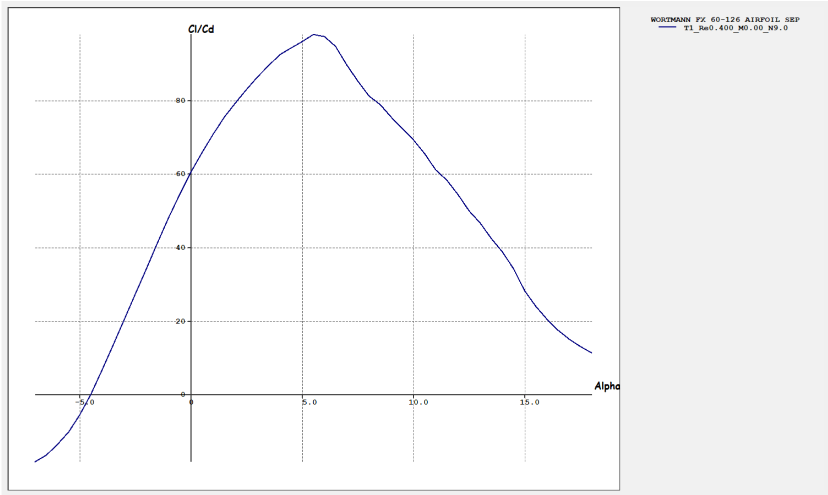


Figure 6.10: C_L/C_D vs α

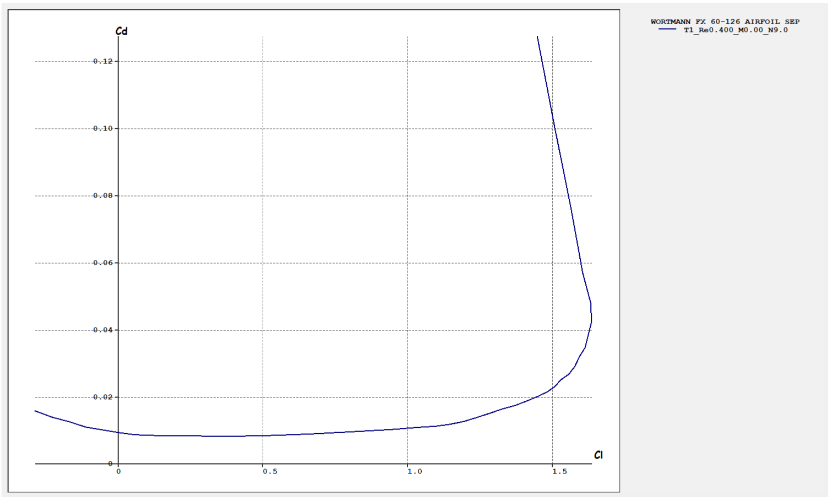


Figure 6.11: C_L vs C_D

6.2 Wing Location

Wing location defines the vertical location of the wing with respect to the fuselage center line. There are 5 types of wing locations/configurations in the case of a monoplane. They are the Low wing, Mid wing, High wing, Shoulder wing, and Parasol wing. Comparing the advantages and disadvantages of these wings and considering our requirements, We have decided to have **High wing configuration**. These reasons for choosing the High-wing configuration are:

1. Eases and facilitates the loading and unloading of payload from the bottom of the UAV, without affecting the wings.

2. Easy to manufacture as the wing can be manufactured as an entire single part and fixed on top of the fuselage instead of fabricating two parts of the wing and connecting them separately to the fuselage.

3. Saves the wing from high-temperature exit gases in a vertical take-off and landing (VTOL) aircraft. The reason is that the hot gases bounce back when they hit the ground, so they wash the wings afterward. Even with a high wing, this will severely

reduce the lift of the wing structure. Thus, the higher the wing is the farther it will be from the hot gases.

4. Increases the dihedral effect $C_{l\beta}$. It makes the aircraft laterally more stable. The reason lies in the higher contribution of the fuselage to the wing dihedral effect ($C_{l\beta_w}$).

5. 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.

6. The aerodynamic shape of the fuselage lower section can be smoother.

7. There is more space inside the fuselage for payload, powerplant, sensors, and other equipment.

8. 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 ($M_{D_{cg}} > 0$)

There are a few disadvantages even in the case of a high wing configuration like:

1. The aircraft tends to have more frontal area compared with mid-wing configuration. This will increase aircraft drag.

2. The wing produces more induced drag (D_i) due to the higher lift coefficient.

3. A high wing is structurally about 20% heavier than a low wing.

4. The aircraft lateral control is weaker compared with mid-wing and low-wing since the aircraft has more laterally dynamic stability.

As we consider these advantages and disadvantages, we can conclude that high wing configuration is the best possible configuration for our requirements.

6.3 Wing design

In this section, we will look into the parameters that will be used for designing the wing.

- **Span area:** From the previous chapters on wing loading we estimated the wing loading to be (W/S = 210 $\frac{N}{m^2}$) taken as an average between the range of wing loading. From previous values we have MTOW = 23.5 kg, for this we get,

$$S = \frac{W}{W/S} = \frac{(23.5)(9.81)}{210} = 1.08m^2 \tag{6.1}$$

- **Wing Span:** From the previous dataset on the VTOLs added in chapter 3 of Drag estimation the values of span for the VTOLs near our MTOW(23.kg) is found to be as 3.6 m **(b = 3.6 m)**

- **Aspect ratio:** From the chosen value of wing span and calculated value of span area we get the aspect ratio from it.

$$AR = \frac{b^2}{S} = \frac{(3.6)^2}{1.08} = 12 \tag{6.2}$$

- **Taper ratio:** The taper ratio selected is based on the span area and aspect ratio, we design the tip chord and root chord of the wing such from that root and tip chord length we get the calculated span area. For optimized values using XFLR5, the values for chord length at root and tip are given **Root chord length = 0.38 m** and **Tip chord length = 0.22 m**. From this, we will get the taper ratio to be λ**=0.58**.

  For the used taper ratio and the aspect ratio we have we could calculate the Ostwald efficiency. From the research article [36] we took the following graph,

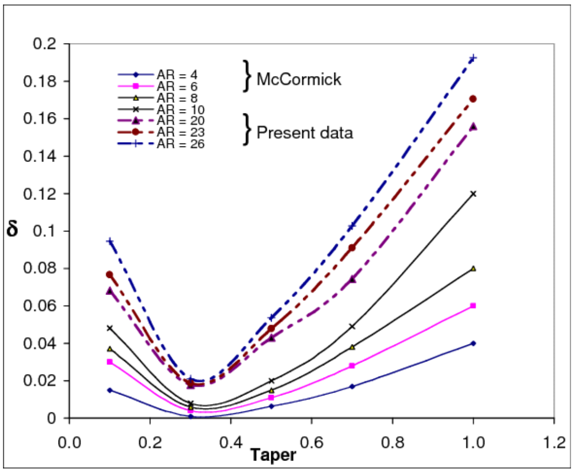


Figure 6.12: Efficiency of tapered wing

  From the graph, even though we don't have data for our aspect ratio as an approximate we see that for our values of taper and aspect ratio we have $\delta < 0.04$. So for this value, we get the Ostwald efficiency factor to be **e = 0.9614** which is pretty good (Close to elliptic wing distribution).

- **Sweep:** Giving a sweep angle for the wing improves static lateral, longitudinal, and directional stability and also improves the aerodynamic characteristics like lift, drag, and pitching moment. However, the sweep angle is disadvantageous in subsonic flight conditions(incompressible flow region) due to the increase in manufacturing cost and complexity. Since our cruising speed falls in the incompressible flow region, we choose a wing with **zero swept angle**.

- **Dihedral angle:** The presence of a dihedral angle in the wing, helps in the increase of lateral stability. But there are disadvantages too such as increased drag, manufacturing cost, the lift produced is not completely vertical so considering these situations we will go with a wing with **no dihedral angle**. Keep in mind that the roll stability coefficient will be provided by considering high wing configuration.

- **Twist:** Since the control surface such as an aileron is added near the tip of the wing, at stalling conditions we prefer that the regions near the tip. So we will consider having a twist at the tip (around 2 degrees) so that when the root is in the stall region the tip is not stalled.

6.4 Design of wing

The wing was designed in XFLR5 software and the aerodynamic coefficient for the wing is plotted and attached. First, let's see the three view diagram of the wing.

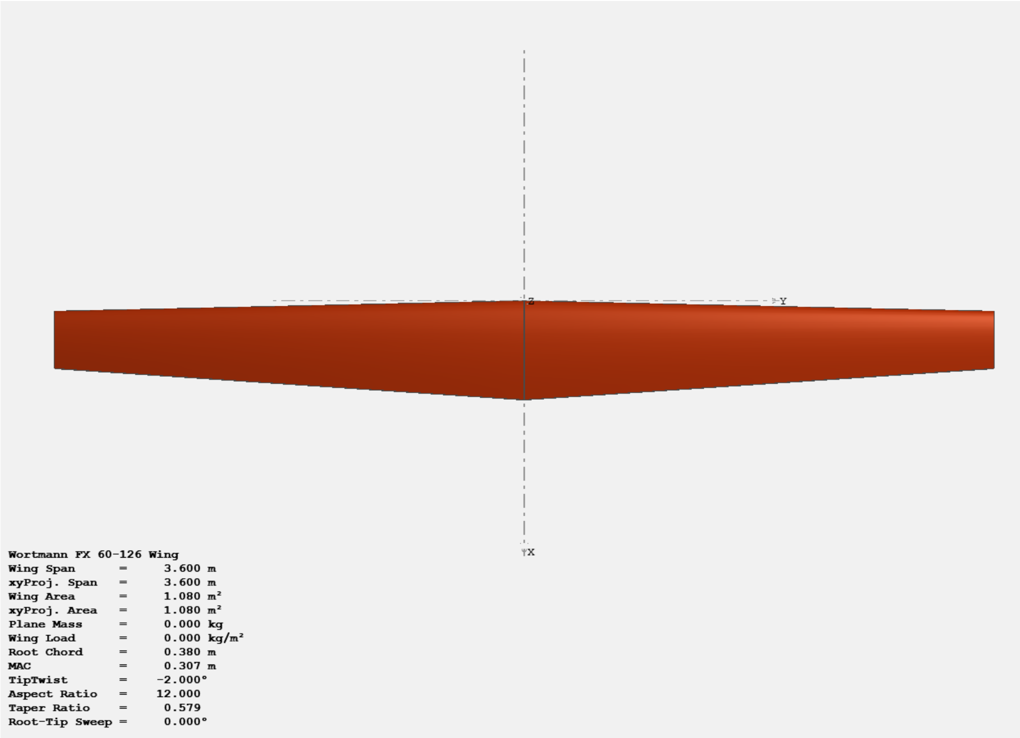


Figure 6.13: Wing view (Top view)

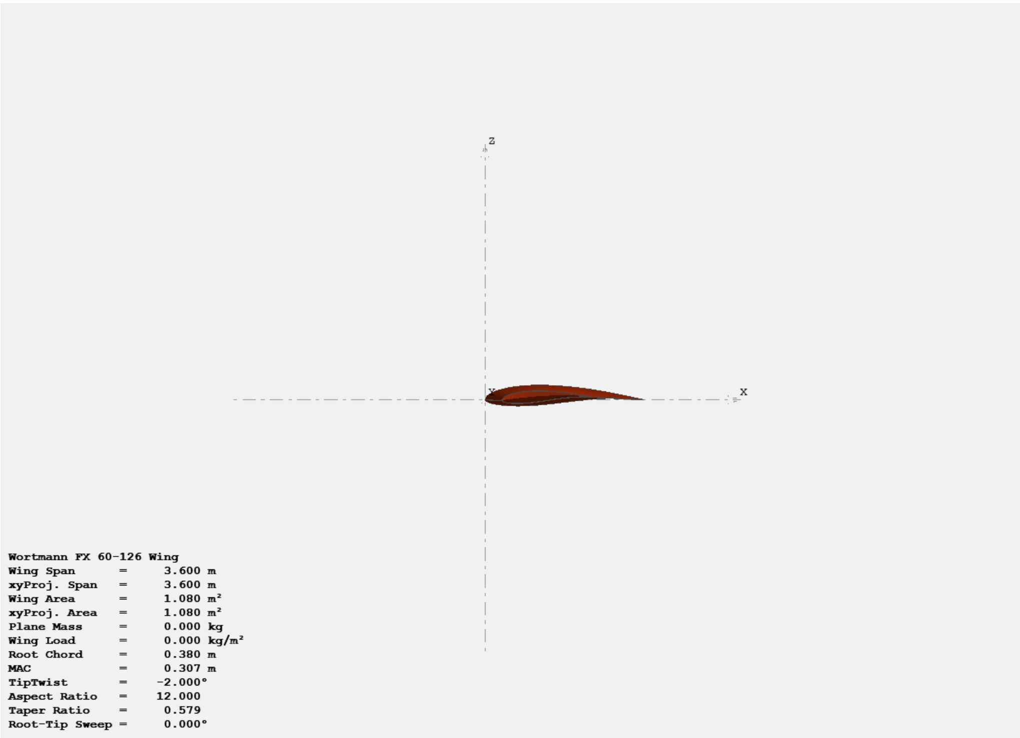


Figure 6.14: Wing view (Side view)

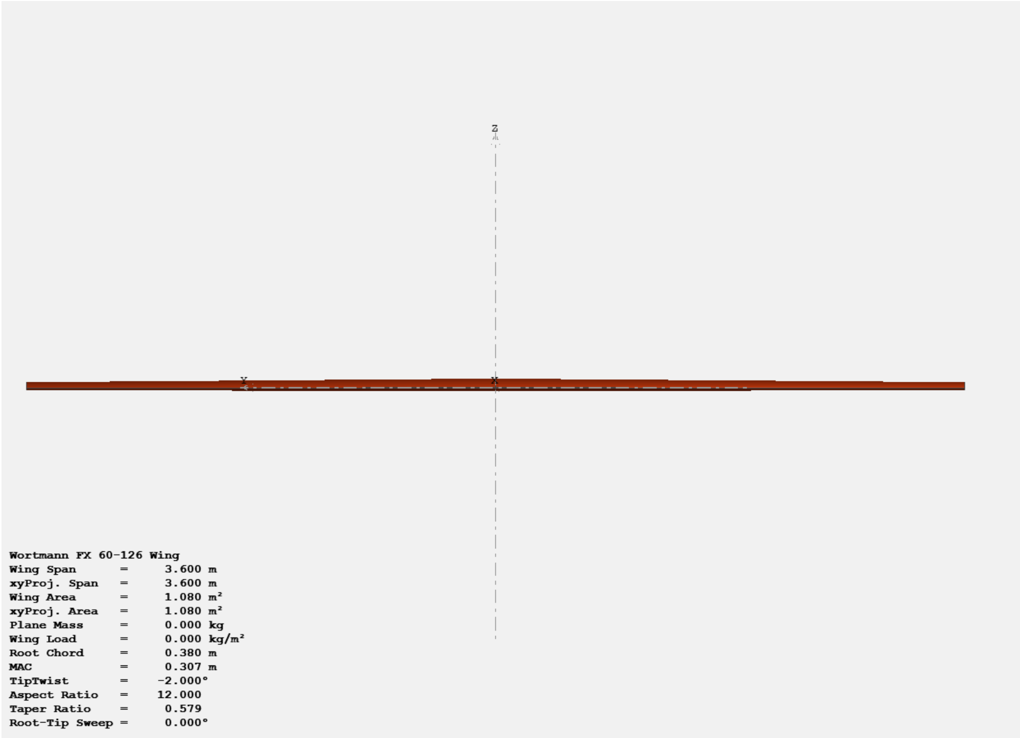


Figure 6.15: Wing view (Front view)

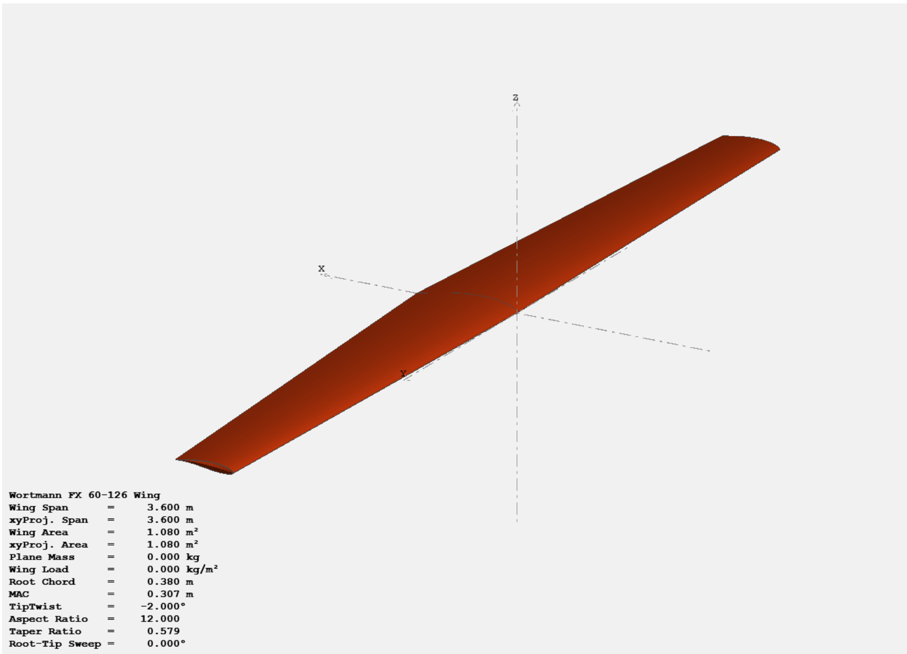


Figure 6.16: Wing isometric view

6.5 Characteristics of wing

In this section, the characteristics of the wing (Aerodynamic coefficients are plotted) using XFLR5 (**The LLT(Lifting Line Theory)** was used to simulate the wing) are attached below,

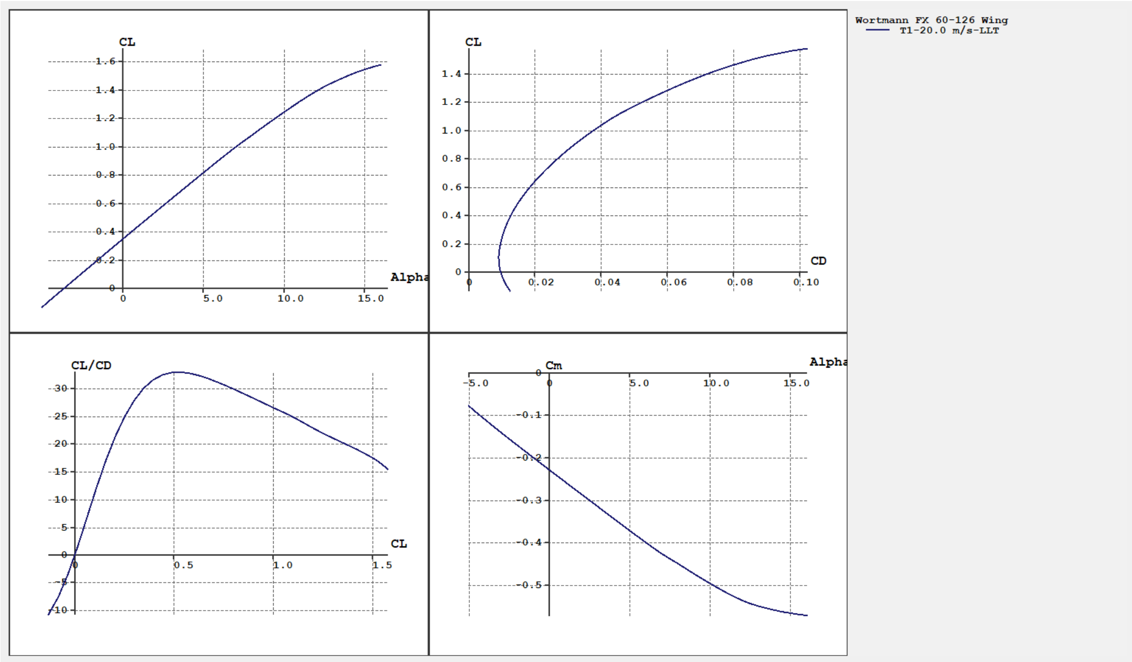


Figure 6.17: Aerodynamic coefficients plot

From the characteristic graph, we can see that the stability derivative C_{m_α} is negative, but the C_{m_0} which has to be positive is not but this will be countered by the contribution from the tail. The lift and drag values at our operation value using LLT (Lifting Line theory) are attached below.

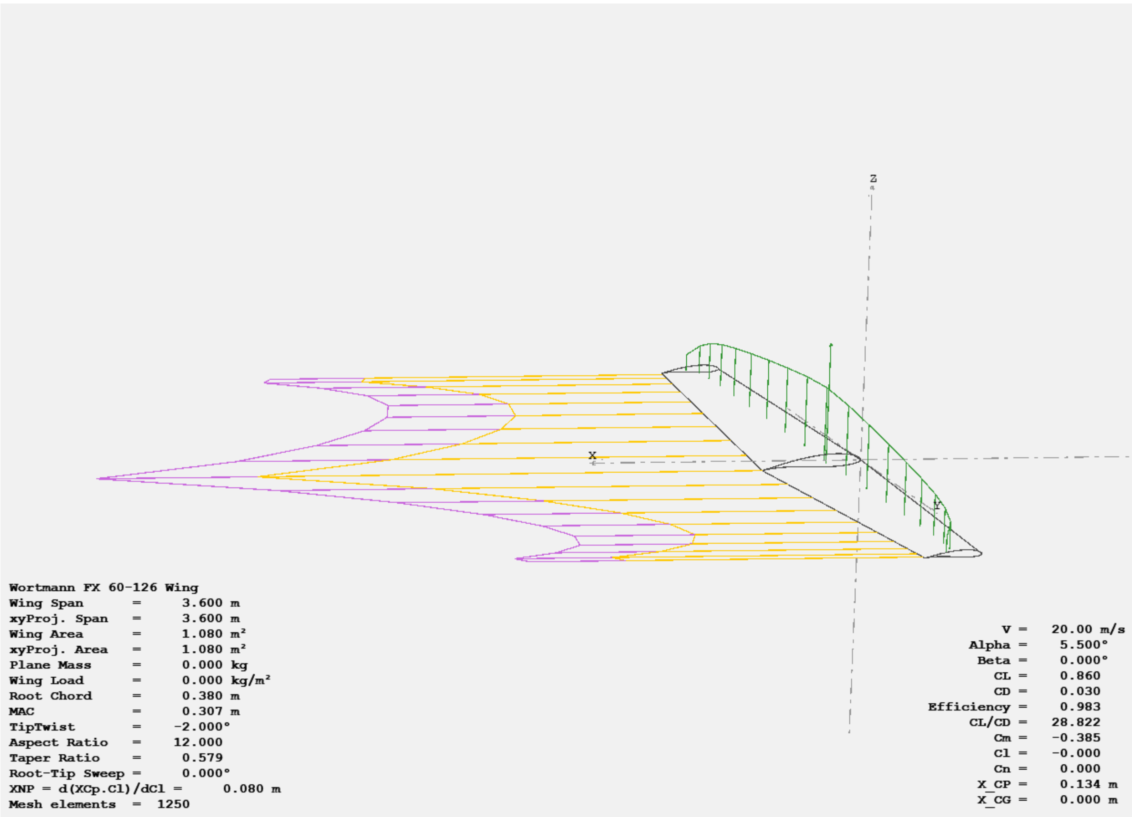


Figure 6.18: Operational point (cruise)

6.6 Aileron area

We will need to select the appropriate specifications for the discrete aileron for roll control. The induced drag, rolling moment, and yawing moment for an aircraft depend in part on the location and size of the ailerons.

Plane flap-type ailerons are the most common ailerons used in fixed-wing VTOLs. Plane flap-type ailerons would have a chord ratio between 10 and 40 percent of that of the wing. Aileron span to the semi-wing span ratio of 30 to 60 percent is the most commonly seen setup.

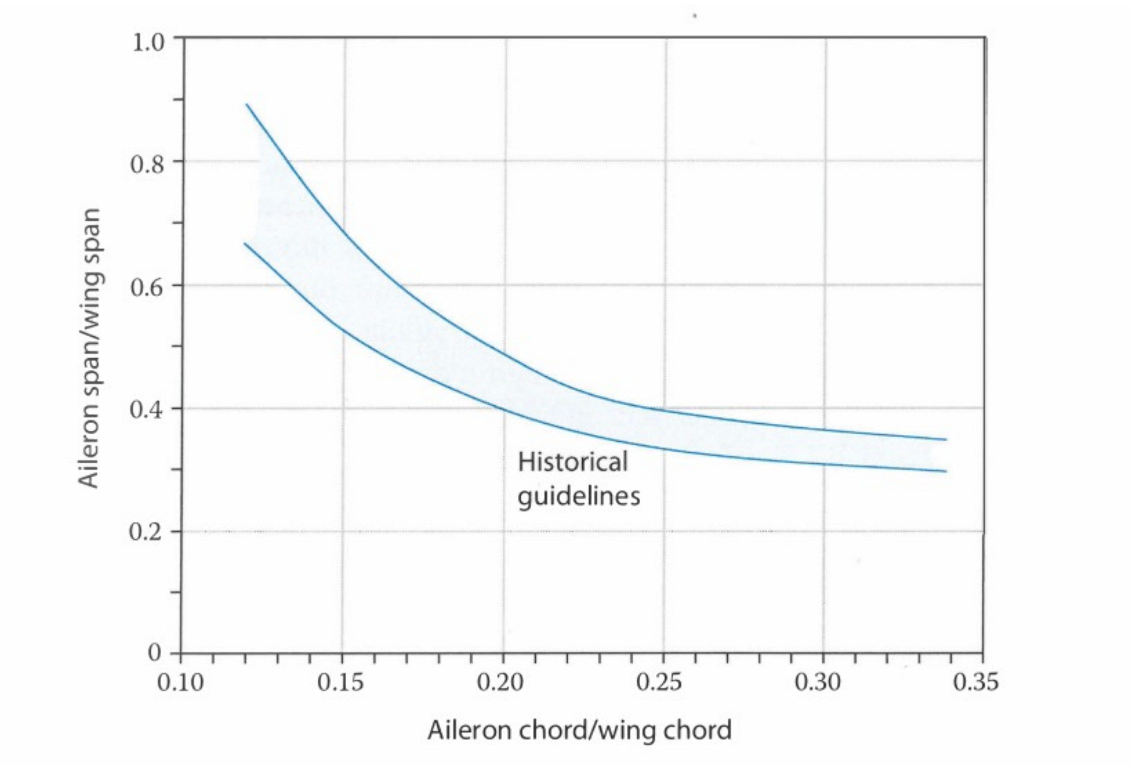


Figure 6.19: This graphic gives us an approximate idea on the ideal ranges of aileron span and chord.

The optimum aileron area is about 10 percent with respect to the semi-wing area. Therefore, the optimized chord length for the aileron is taken to be 25 percent of the wing chord and the aileron span is 40 percent of the semi-wing span. The below-attached figure is taken from D. Raymer's book [31] Chapter 6 (Fig:6.3) which gives the aileron sizing guidelines.

$$\frac{C_{aileron}}{C_{wing}} = 0.2 - 0.25$$

$$\frac{b_{aileron}}{b_{wing}} = 0.2 - 0.4$$

$$\frac{S_{aileron}}{S_{wing}} = 0.04 - 0.1$$

In the above conditions, b_{wing} and S_{wing} are the span and area, respectively for the semi-wing and not the entire length of the wing. C_{wing} is the chord length for the wing and $C_{aileron}$ is the chord length for the aileron.

Finally, we take

$$\frac{C_{aileron}}{C_{wing}} = 0.2$$

$$\frac{b_{aileron}}{b_{wing}} = 0.4$$

$$\frac{S_{aileron}}{S_{wing}} = 0.08$$

6.7 Wing incidence angle estimation

The angle between the chord line of the wing and the fuselage reference line is known as the Wing incidence angle. It depends on the required cruise lift coefficient and zero lift angle of attack of the airfoil. It can be determined using the following equation,

$$i_w = \alpha_0 + \frac{C_{l_{cruise}}}{C_{l_\alpha}}$$

where,
α_0 = -3.64°
$c_{l_{cruise}}$ = 0.857
c_{l_α} = 0.092 /deg
These are found in the polar plotted in the design of the wing section.

$$\Rightarrow i_w = 5.6^\circ$$

According to XFLR simulations, the wing has the required characteristics at 5.5°. Hence the aircraft will be almost parallel to the ground in cruise conditions.

6.8 High lift devices

Since our required value of $C_{L_{max}}$ is attained by the wing before stall we do not want to add high-lift device. Also adding a high lift device is not favorable in low Reynolds number flows, as they will suffer from laminar stall. So we will not be considering adding high lift devices such as flaps etc.

6.9 Winglets

Winglets could help in reducing the induced drag due to vortices trailing at the tip of the airfoil, with a small increase in profile drag. Adding winglets would also make the analysis complex also reduce the manufacturing feasibility. Since we have enough power from the motor to overcome the drag, we are not considering winglets.

6.10 Collection of parameters used

So the parameters used, from the previous section that we used for the wing is as follows,

Table 6.3: Wing Design parameters

Parameters used	Values used
Span area	1.08 m^2
Wing span	3.6 m
Aspect ratio	12
Taper ratio	0.58
Root chord	0.38 m
Tip chord	0.22 m
Sweep	0 degree
Dihedral angle	0 degree
Twist	-2 degrees (at the tip)
Aileron area to span area	0.08
Wing incidence angle	5.6 degrees

Chapter 7

Midsem Presentation Queries

7.1 Problem Encountered

Our configuration of lift+cruise is popular for its inherent redundancy and the option to tailor separate propulsion systems for each flight regime, However during cruise, the vertical flight propellers of a lift+cruise design are inactive and exposed, this increases the projected area of a body which ultimately leads to additional drag.

In this chapter, we specify the suitable orientation for our vertical propellers that will lead to minimal drag.

Initially, we rule out the following possibilities used to encounter the drag:

7.1.1 Variable pitch rotor

We avoid using a variable pitch rotor due to:

- Complexity in Manufacturing
- It is generally used in larger aircraft and we are designing a small-scale UAV.
- Fixed pitch propeller reduces the number of propellers used.

7.1.2 Retractable Propeller Mechanisms

We avoid using a retractable propeller mechanism due to:

- Contribution to additional mass and cost.
- Additional Complexity and reliability issues

We use an intermediary solution that is employed on most crewed electric VTOL aircraft which requires the use of the motor controller to align the propellers to a specific orientation which is usually parallel to the slipstream.

7.2 Research Review for small scale UAV

According to a research review the observations considered are:

- No significant change in the lift coefficient was observed when varying the stopped azimuthal angle of the propellers. (This suggests that flow exiting the stopped front propeller does not significantly disturb the flow across the downstream wing)
- Up to a 30% decrease in drag is observed when the propellers are parallel to the wing chord line as compared to when both propellers are aligned perpendicularly.

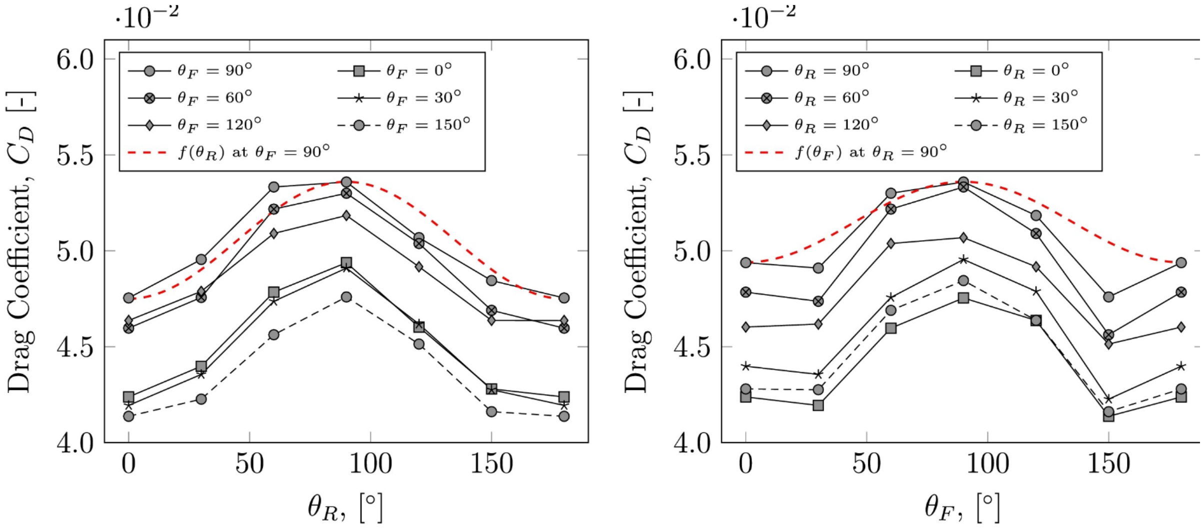


Figure 7.1: a) Image shows the drag variation with azimuthal angle for rear propellers, b) Image shows the drag variation with azimuthal angle for front propellers.

- The stopped azimuthal angle of the front and rear propeller could result in a 36% variation in the lift-to-drag ratio of the model.

7.3 Conclusion

- Three factors affect the amount of drag produced which are propeller diameter, blade pitch and airspeed
- The orientation for the propeller should be along the streamline direction.
- The motor will be equipped with a self-locking mechanism to restrain the free motion of the propeller.

7.4 GPS for locating the fixed Wing VTOL

7.4.1 Problem

Since the VTOL will be required to make turns in case of an obstruction, we would need an active location of the UAV which can be done through a GPS along with a long-range transmitter like LoRa (Long Range) which gives a range of 20kms or SigFox which can give a range of around 50kms. The GPS module that can be used is :

7.5 Transitioning of the VTOL

7.5.1 Problem

How does the VTOL transition from the vertical thrust-producing thrusters to the cruise?

- After the desired altitude is reached the vertical thrusters are deaccelerated and the cruise propeller is simultaneously accelerated to generate the lift which counters for the loss in vertical thrust such that the altitude is maintained.

- 1 x Neo 6M GPS Module

Neo 6M GPS Module Specifications:

Input Voltage	2.7 - 3.6V
Model	GY-GPS6MV2 Ceramic antenna
Mounting Hole Diameter	3mm
Baud Rate	9600 Baud default (Configurable from 4800 to 115200 Baud)
Antenna size	18 * 18mm
Module size	23mm * 30mm
Cable	20mm
Weight	50 grams

Figure 7.2: L86 GPS module

Figure 7.3: SigFox Module for long range Transmission

$$W = L + T \tag{7.1}$$

where W is the weight of the quadplane, L is the lift generated due to wings and T is the reducing vertical thrust. **The rate of lift generation = The rate of decrease of vertical thrust**. The vertical motors will be running until the stall condition($C_{L_{max}}$) is attained.

Chapter 8

Fuselage Design and Tail Layout

In this chapter we will design the fuselage of our VTOL and look into the aspects and configurational basics of the tail of our VTOL. For this part we will look into the historical data and also references to obtain the parameters needed for design. A detailed analysis is the chapter to follow.

8.1 Historical data for fuselage

In this section we will look into the historical data collected to find the optimal diameter and length of the fuselage.

Table 8.1: Historical data for fuselage

Name	MTOW (Kg)	Fuselage Max diameter (cm)	Length of fuselage (cm)
TEKEVER	25	30.163	170
FlyDragon/ Baby shark 220 VTOL	12	20.4866	87.9524
Yangda / FW-250	13	22.2354	91.157
FlyDragon FDG410 VTOL	30	26.1322	192.6
Foxtech Great Shark 330	23	25.7801	110
Foxtech AYK-250 Pro VTOL	15.5	21.9339	126
Foxtech AYK-350 VTOL	35	29.4056	188
Flydragon /FDG33 VTOL	18	23.32	150

From this historical data we have, the decided **diameter of the fuselage is 23cm** (such that the payload diameter is around 25cm). The corresponding chosen **fuselage length is 1.6 m**. Considering VTOLS closer to out MTOW and closer to our payload weight.

8.2 Parameters required for fuselage design

The preliminary parameters needed for fuselage design are decided and are added as shown in this section.

Table 8.2: Parameters of fuselage design

Parameters Used	Value of the parameter
Length of the fuselage	1.6 m
Maximum diameter fuselage	23 cm
Dimension of battery	25.6 * 19.3 * 8.0 cm^2
Dimension of payload	ϕ = 20 cm ; L=52 cm
Rounded edge of nose	ϕ = 3.2 mm (motor shaft)

8.2.1 Fuselage dimensions

From the value of length of the fuselage and the maximum diameter of the fuselage the ratio of length to the diameter (here maximum diameter) is,

$$\frac{L}{D_{max}} = \frac{160}{23} = 5.925 \tag{8.1}$$

From the M.Sadarey textbook[26], Chapter 7 (eq 7.8a) the optimal L/D_{max} ratio for minimum parasite drag was found out to be **5.1** . But here we have 5.925 as we have a large diameter owing to the payload carrying capacity that is required for delivery. As we are concerned with the parasite drag (minimal) here, the change in parasite drag was found not to change significantly.

8.2.2 Battery dimensions

The battery chosen for the VTOL is Mpower battery of capacity 45000 mAh , configuration as 12S with 44 Volts. The battery was found to have dimension of **25.6 cm** * **19.3 cm** * **8 cm** as length, width and height respectively.

Figure 8.1: Battery used

8.2.3 Payload dimensions

The payload dimension is chosen accordingly such as since the maximum diameter of the fuselage is chosen to be 23 cm , we will choose the payload diameter to be around 20 cm as the 3cm is given as a structural space. The payload carrier is a cylinder, with a diameter of 20 cm and a length of around 53 cm. Lets see how this was chosen,

This was because from the reference to Raymer's book [31] Chapter 8 (Fig:8.2) the upsweep angle should be less than 12 degrees to reduce the flow separation and its effects. This is shown as an image as follows,

By considering this upsweep angle and since we know the dimensions of the battery after which we will place the payload. Since our payload dimensions are not fixed, from the chosen length and diameter of the wingspan we see that the portion of the fuselage with maximum diameter after the battery is of length 53 cm .

Figure 8.2: Upsweep angle

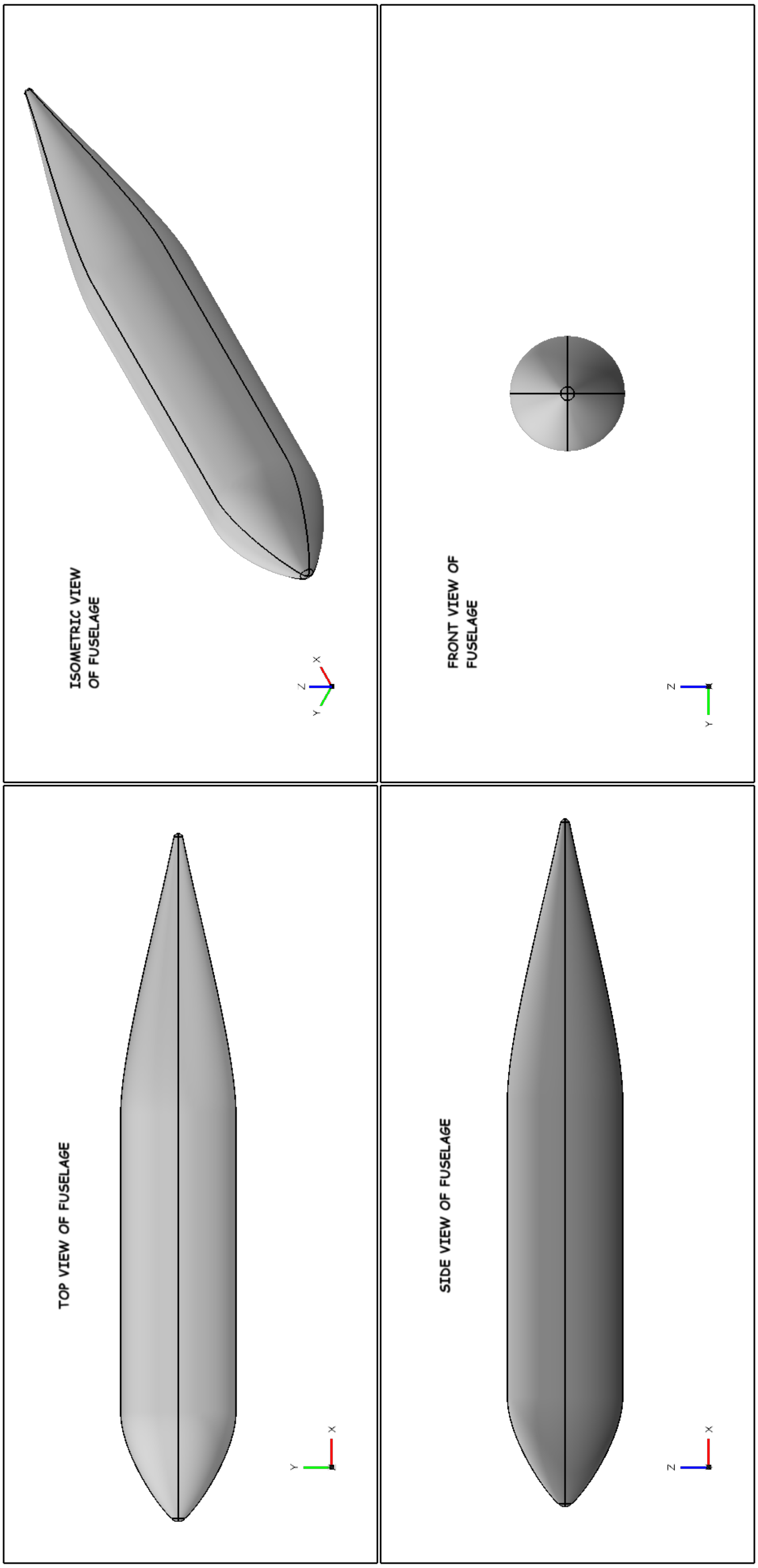


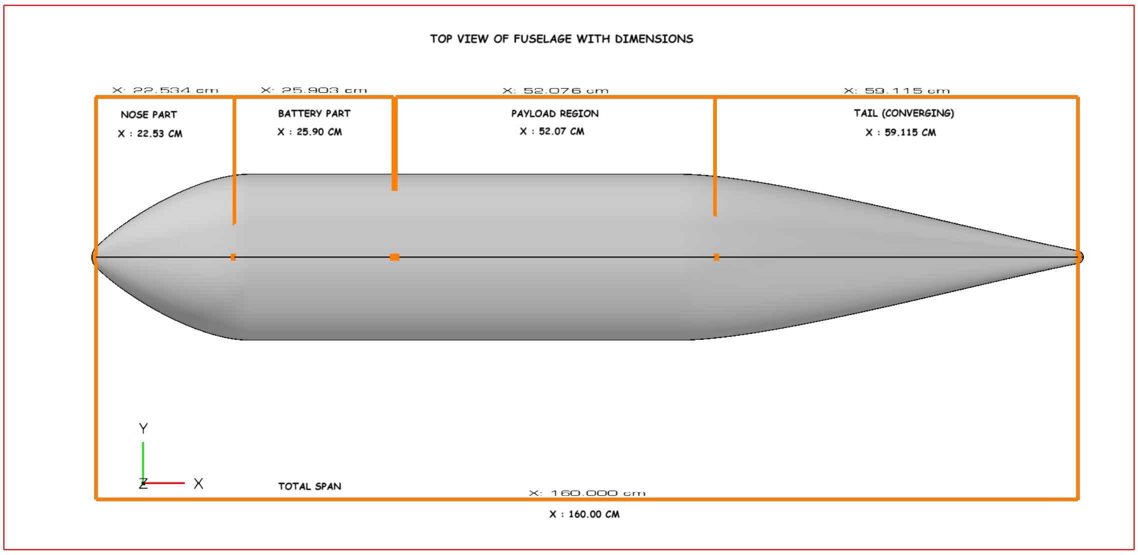
Figure 8.3: Views of the fuselage design

8.3 Design of fuselage

In this section we will see the different views of the fuselage with dimensions and view on how we place our items on the fuselage.

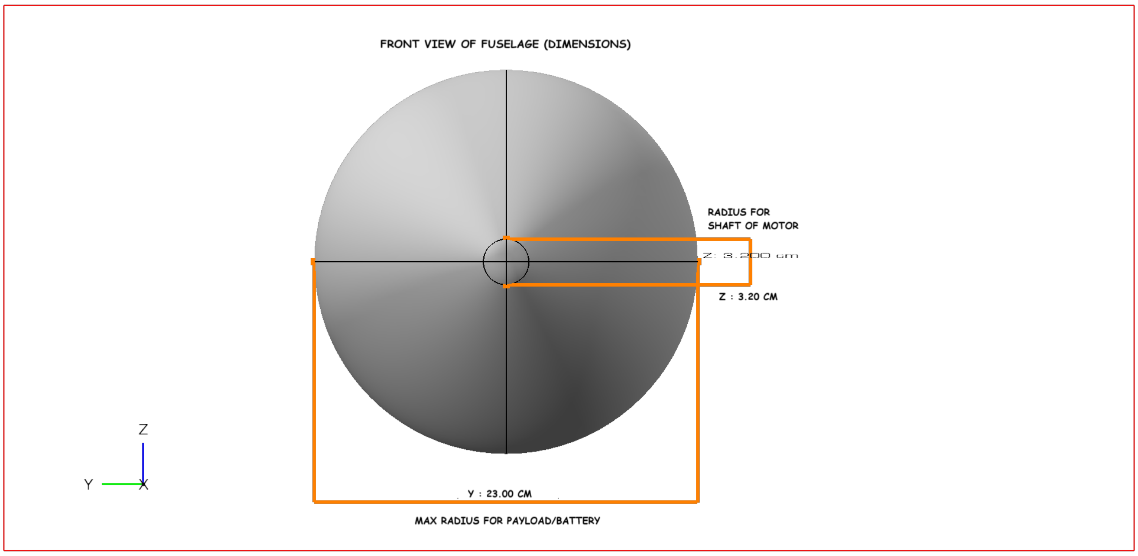
8.3.1 Top view of fuselage

In this subsection we will see the top view with marked dimensions, In the figure attached below, the marked parts denote which object will be placed in that region.



8.3.2 Front view of fuselage

In this subsection we will see the front view with marked dimensions, In the figure below , radius of the shaft marked here is taken from the dimensions of motor.



8.4 Previous Data on Tail Parameters

Data collected from the previous aircraft for the horizontal tail:

Table 8.3: Historical data for Horizontal tail

Parameters	Flydragon FDG 410 VTOL	CUAV Raefly VT260	Flydragon fdg33 vtol	AYK350	AYK 250 Pro
Configuration	Conventional	Conventional	Conventional	T Tail	T Tail
Span	0.855	0.754	0.725	0.964	0.6617
Root chord	0.287	0.229	0.212	0.398	0.2166
Tip chord	0.1137	0.1393	0.1304	0.2037	-
Taper Ratio	0.407	0.234	0.615	0.298	0.271
Sweep Angle	18	10	18.58	-	-
Tail Arm	0.841	0.898	0.859	1.015	0.756

Data collected from the previous aircraft for the Vertical tail:

Table 8.4: Historical data for Vertical tail

Parameters	Flydragon FDG 410 VTOL	CUAV Raefly VT260	Flydragon fdg33 vtol	AYK350	AYK 250 Pro
Configuration	Conventional	Conventional	Conventional	T Tail	T Tail
Span	0.375	0.283	0.247	0.295	0.173
Root chord	0.303	0.263	0.219	0.259	0.18
Tip chord	0.102	0.132	0.144	-	-
Taper Ratio	0.499	0.461	0.661	0.750	0.844
Sweep Angle	18	21	26.7	-	-
Tail Arm	1.041	0.847	0.835	1.145	0.644

8.5 Tail Configuration

Previous data mostly used conventional, T-tail and V-tail type of configurations. For our VTOL, We have decided to go with the conventional tail design type due to following reasons:

Conventional Tail Type:

- In a conventional tail, the horizontal stabilizer is mounted low on the vertical stabilizer, usually at the base.
- Though it has reduced efficiency due to the horizontal tail lying in the wake region, it provides appropriate stability and control.
- It leads to a compact and lightweight construction.

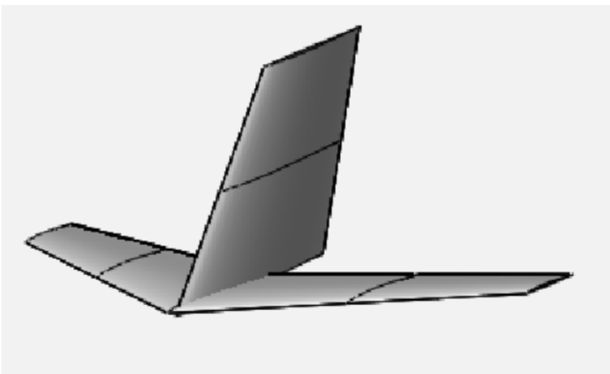


Figure 8.4: Conventional tail

T tail Type:

- A T-tail configuration mounts the horizontal stabilizer on top of the vertical stabilizer, creating a T shape when viewed from the front.
- This design moves the horizontal tail completely out of the engine wake, improving its effectiveness.

- It results in additional weight and structural complexity
- It increases the possibility of deep stall which occurs in tail.

V-tail

- V-tail replaces the traditional fin and horizontal surfaces with two surfaces which forms a V-shape configuration.
- Due to reduced intersection surfaces from three to two, it leads to the reduction in drag as it eliminates interference drag.
- V-tail aircraft need to have a longer rear fuselage compared to the conventional empennage, To prevent snaking/yawing.
- It has a Complex control system and requires more control forces to effectively operate the aircraft.
- It has poor failure tolerance and so difficult to maneuver the aircraft if one of the lifting surfaces has issues or malfunctions.

Thus taking account of advantages and disadvantages of above tail configurations: **We will be going with the convention tail type.**

8.6 Horizontal Tail Parameters

In this section we will estimate the tail parameters as a preliminary design which will be used in next chapter for detailed analysis.

8.6.1 Airfoil Type

For both horizontal and vertical tails we will be going with a symmetric airfoil as it doesn't produce lift at zero angle of attack and so minimizes induced drag. From M.Sadraey [26], Chapter 6,it was given that NACA0009 and NACA0012 as the most used airfoils for this purpose. Further selection will be done based on stability analysis in the tail.

8.6.2 Tail Arm

According to a reference in Raymer [31], the tail arm for an aircraft with a front-mounted propeller is 60% of fuselage length. Given our fuselage length as 1.6, we get:

$$L_{HT} = 0.6(1.6) = 0.96m \tag{8.2}$$

There was also another formula from M.Sadraey [26]. Chapter 6 (Eqn 6.47) for finding the optimum tail arm reducing drag and it was found with the historical data to match with the above approximation.

8.6.3 Horizontal Tail area

The horizontal tail volume coefficient V_{HT} for our aircraft according to typical values & reference in Raymer [31] Chapter 7, can be approximated as **0.5** .

$$S_{HT} = \frac{V_{HT} C_w S_W}{L_{HT}} \tag{8.3}$$

Here,
C_w is the Mean Aerodynamic chord of the wing (0.307 m).
L_{HT} is the Tail arm calculated in the previous section.

$$S_{HT} = \frac{(0.5)(0.307)(1.08)}{0.96} \tag{8.4}$$

$$\boxed{S_{HT} = 0.172m^2} \tag{8.5}$$

8.6.4 Horizontal Aspect ratio

From the reference M.Sadraey [26], Chapter 6 (sec. 6.7.5) we get the following expression as an approximation,

$$AR_T = \frac{2}{3}(AR) \tag{8.6}$$

$$AR_T = \frac{2}{3}(12) = 8 \tag{8.7}$$

The **Aspect ratio for horizontal tail is 8** .

8.6.5 Span of horizontal tail

From the aspect ratio and the span area we could get the span of the horizontal tail

$$AR_T = \frac{b_{HT}^2}{S_{HT}} \tag{8.8}$$

$$b_{HT} = 1.1m \tag{8.9}$$

The **Span for horizontal tail is 1.1 m** .

8.6.6 MAC of horizontal tail

From the aspect ratio and the span of the horizontal tail we could get the MAC of the horizontal tail.

$$AR_T = \frac{b_{HT}}{c_{HT}} \tag{8.10}$$

$$c_{HT} = 0.1375m \tag{8.11}$$

The **Mean aerodynamic chord for horizontal tail is 0.1375 m** .

8.6.7 Taper ratio of horizontal tail

From the Sadraey [26], Chapter 7 (sec 6.7.6) we could see that taper ratio criteria of wing which makes the wing lift distribution close to elliptic one. This requirement is not needed to be satisfied at the tail. Thus for now we will consider that taper ratio as 1. Only consideration will be to reduce the tail weight (But this might change later depending on stability analysis).

8.6.8 Sweep, twist and dihedral angle of horizontal tail

From the data we have till now, we do not consider any of twist, dihedral and sweep effects. These effects might come into play when we consider stability analysis.

8.6.9 Tail incidence angle

The wing incidence angle for the horizontal tail is not found with the data we have till now and will be done with once we have the location of center of gravity of the aircraft.

8.7 Vertical tail parameters

In the section similar to previous section we will find the parameters for analysis later.

8.7.1 Airfoil Type

Similar reason to the airfoil type selection given in horizontal tail part.

8.7.2 Vertical Tail area

The vertical tail volume coefficient V_v for our uav can be approximated as 0.03 according to M.Sadarey [26] reference as it says volume coefficient can be around 0.02 - 0.12. From the previous historical data we approximated it to be **0.03**.

$$S_{VT} = \frac{V_v bw S_W}{L_{VT}} \tag{8.12}$$

$$S_{VT} = \frac{(0.03)(3.6)(1.08)}{0.96} \tag{8.13}$$

$$S_{VT} = 0.1215m^2 \tag{8.14}$$

8.7.3 Vertical tail Aspect ratio

The average span for vertical tail calculated from previous data is around **0.6 m** (Nearly similar configuration and MTOW) and the area is calculated above.

$$AR_v = b^2/S_{VT} = 2.96 \tag{8.15}$$

8.7.4 Mean aerodynamic chord length

From the previous data of span and aspect ratio we will get the MAC of the vertical tail

$$AR = b_{VT}/c_{VT} \tag{8.16}$$

$$c_{VT} = 0.2025m \tag{8.17}$$

8.7.5 Sweep of vertical tail

From the Sadraey reference [26] we see that the tail sweep angle should be less that 20 degrees. As the sweep angle of the vertical tail is increased, the yawing moment arm is increased which improves the directional control of the aircraft. Subsequently, an increase in the vertical tail sweep angle weakens the aircraft directional stability, since the mass moment inertia about the z -axis is increased. Further analysis of the sweep angle will be determined in stability analysis.

8.7.6 Other missing parameters

The other missing parameters such as Taper ratio will be done after the stability analysis is done. As the taper ratio has effects on the location of aerodynamic center on the vertical tail.

8.8 CAD model of tail

From the estimated values of the parameters of tail (The dimensions are present in the previous sections), the CAD model of the tail is attached below,

Figure 8.5: Front view of tail

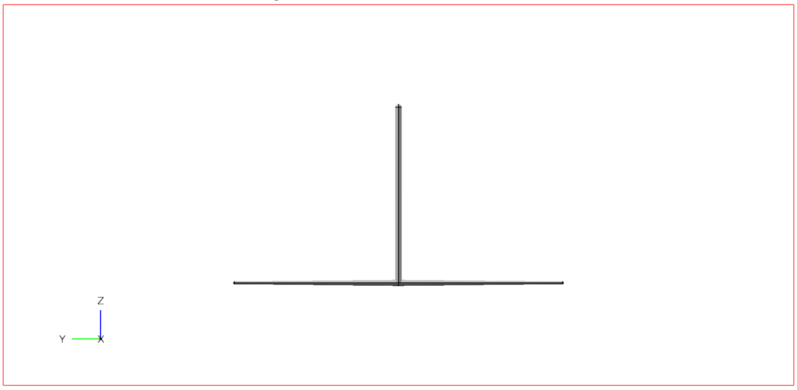


Figure 8.6: Left view of tail

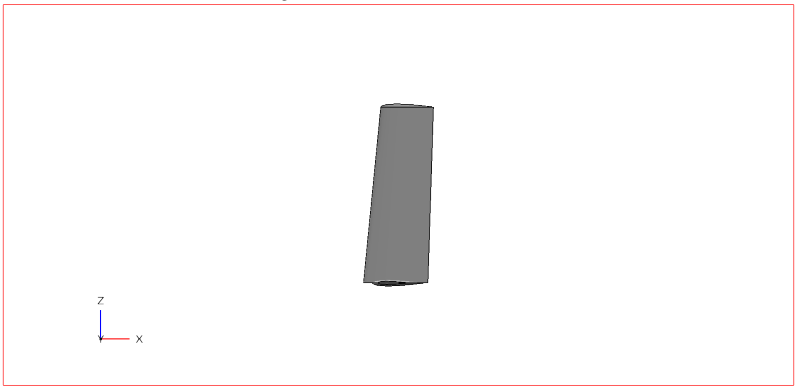


Figure 8.7: Top view of tail

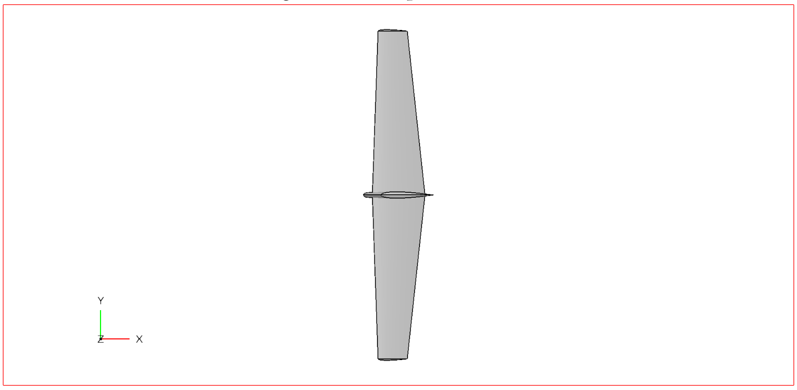
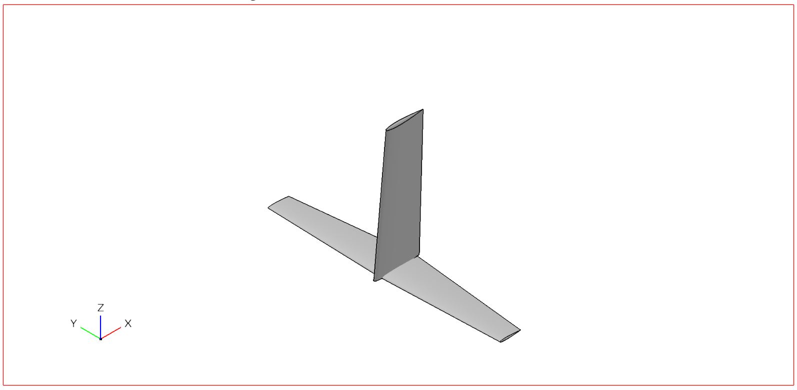


Figure 8.8: Isometric view of tail



8.9 Control Surfaces

The dimensions for elevator and rudder control surfaces from previous aircrafts is mentioned below:

Parameter	Flydragon FDG 410 VTOL	CUAV Raefly VT260	Flydragon fdg33 vtol	Average
Rudder Length	0.284	0.2279	0.1715	0.2279
Rudder Width	0.098	0.0483	0.0464	0.0643
Elevator Length	0.284	0.2907	0.608	0.394
Elevator Width	0.069	0.0466	0.0464	0.054

Table 8.5: Historical data off the control surfaces

8.9.1 Control Surface Sizing

The primary control surfaces for the horizontal and vertical tail are the elevator and rudder respectively. From the [26] chapter 12 The area ratio of elevators to the horizontal tail span area is 20-35% and Rudder is 15-20% of the vertical tail area.

Area for Elevators:

$$Area = (0.3)(0.172) = 0.0516m^2 \tag{8.18}$$

Area for Rudder:

$$Area = (0.15)(0.1215) = 0.018m^2 \tag{8.19}$$

From M.Sadraey ([26]), Chapter 12 we get to decide the following parameters for elevator,

$$\frac{b_e}{b_{ht}} = 0.9.$$

$$\frac{c_e}{c_{ht}} = 0.35.$$

Thus span of elevator(b_e) = 1.0 m and chord of elevator(c_e) = 0.05 m.

for rudder,

$$\frac{b_r}{b_{vt}} = 0.6.$$

$$\frac{c_r}{c_{vt}} = 0.3.$$

Thus span of rudder(b_r) = 0.3 m and chord of rudder(c_r) = 0.06 m.

Chapter 9

Landing Gear Design

9.1 Landing gear configuration

We have decided to go with a landing gear instead of a slid configuration for landing as for our VTOL dimensions the slid would add more weight and drag. Thus a landing gear with a chosen configuration suits our mission profile better where weight and minimum drag are crucial requirments. For our application of vertical landing and vertical take-off we need a landing gear mainly to give our vtol the ground clearance required for the cruise propeller and the configuration which can give minimum drag since there is no retraction mechanism. We have considered the following landing gear configurations:

9.1.1 Tricycle landing Gear

It consists of rear wheels located at rear of UAV and a single nose wheel at front.the pros and cons for this configuration are:

- UAVs is less likely to tip over on their nose, this adds to their ground stability.
- Easy access to fuselage and cabin.
- On propeller aircraft it does not provide much ground clearance, so not suitable for unprepared runways.

9.1.2 Taildragger landing Gear

This type of landing gear utilizes two main wheels located ahead of the center of gravity to support the majority of the aircraft's weight and a smaller wheel at the tail.

- Their performance on unimproved runways, the design is suited for uneven and grassy fields.
- Taildragger gear provides more propeller clearance for forward fuselage, has less drag and weight.
- Also helps in the longitudinal stability of the VTOL while flying.

9.1.3 Tandem Landing Gear

This type of landing gear features the main gear and tail gear aligned on the airplane's longitudinal axis.

- allows for the use of flexible wings as well as reduced drag during deployment.
- ideal for aircraft that require a slim profile for aerodynamic efficiency or for fitting into narrow spaces

- ground handling tricky, as there's a tendency for the aircraft to tip to one side or the other.

Considering the above advantages and disadvantages and according to our requirement we have decided **Tail Dragger** as our Landing Gear Configuration. (Considering some of the facts that we don't need wheels as this is a vertical landing, no taxing will be done by the VTOL). Also a modification to the tail dragger will be carried out (the landing gear near the tail will be the same height as the front landing gear). This is to keep the VTOL's fuselage horizontal.

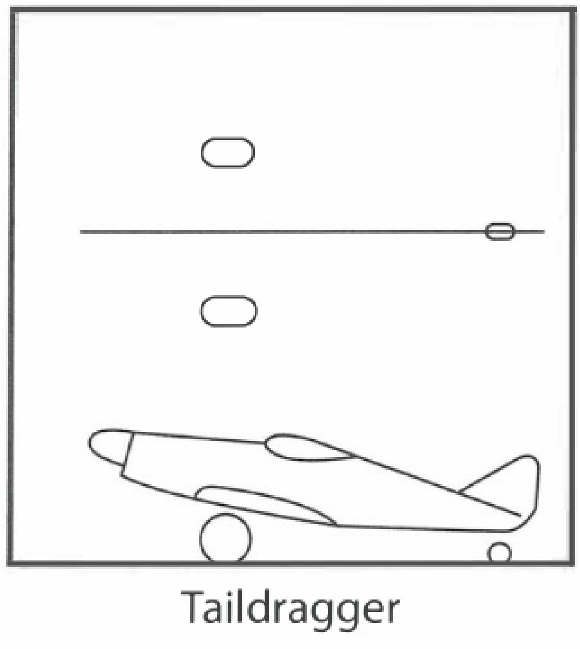


Figure 9.1: Tail dragger configuration

9.2 Impact load on Landing Gear

The landing gear is assumed to land between 1.5 and 4 m/s. The impact force is calculated using the Impulse momentum Equation.

$$F\delta t = mV_f \tag{9.1}$$

where F is the Impact force in Newtons, δt is the impact time assumed as 0.5s (This is taken as such because as the VTOL is very close to the surface the velocity is reduced / Vertical propellor speeds are increasing),m is the mass of the UAV and V_f is the landing velocity.

Velocity (m/s)	Impact Force (N)
1.5	70.2
2	93.6
2.5	117
3	140.4
3.5	163.8
4	187.2

Table 9.1: Velocity and Impact Forces

As we will be landing with the velocity 2 m/s , our impact loading is 93.6 N .

9.3 Determining the location of the Landing gear

For determining the locations of the placement of landing gear, we need to approximate the location of CG, which is done as follows,

Components placed	Mass (kg)	Location of the CG of object
Battery	8 kg	23 cm - 48 cm (COM : 35.5 cm)
Payload	4 kg	48 cm - 103 cm (COM : 75.5 cm)
Wing	2kg	54 cm - 80 cm (COM : 64 cm)
Tail	1kg	COM : 160 cm
Fuselage	2 kg	COM : 75 cm
Powerplant	4kg	COM : 80 cm

Table 9.2: Tentative placement and weight of components

There were some approximations made on the weight of the components which will be analysed in greater detail next week. So some of the approximations made are, the location of the wing is at 40 percent of the fuselage length (From last weeks report tail arm is at around 60 percent of fuselage length [31] Raymer Chapter 7). The fuselage weight and CG location was found from the Fusion 360 software with aluminum alloy 6061 alloy with a thickness of 0.5 mm (manufacturable range starts from 0.03 mm).

Power plant weight/CG on the other hand was found by tentatively placing the four vertical propellers symmetrically and horizontal propeller on the nose of the fuselage. These are approximations which will be further analysed in more detail in further reports.

9.3.1 Determination of the CG

From the table in this section (Tab.9.2), we could find the CG easily. The CG (tentative) for the VTOL with payload is found to be as,

$$CG_{inc.p} = 64 \text{ cm}$$

The CG (tentative) for the VTOL without payload is found to be as,

$$CG_{exc.p} = 61.294 \text{ cm}$$

9.3.2 Distribution of loads

From M.Sadraey [26] Chapter 9 (Sec. 9.3), for our landing gear configuration, we have chosen the location of the landing gear in the following ways,

- The front landing gear components will have to take about 80 % to 90 % of the total load. The rear landing gear component will have to take 10 % to 20% of the total load.
- The front landing gear will be placed in a location in front of the most forward CG, also on the fuselage where it is at maximum diameter.

By considering the above we will choose one such configurations as shown in the figure below,

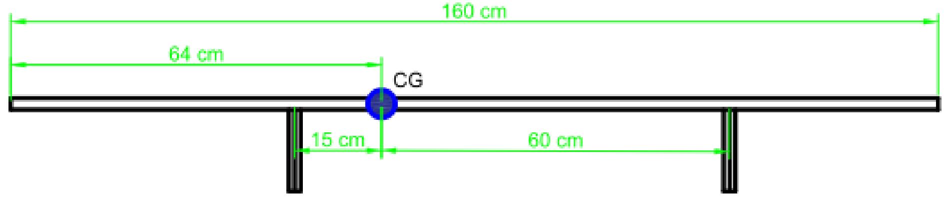


Figure 9.2: Landing gear location with payload

The front landing gear will take 80 % of the load and the rear landing gear will take 20 % of the load. This could be seen from the moment balance equation (about CG) as below,

$$(15)F_{front} = (60)F_{rear} \tag{9.2}$$

Thus,

$$F_{front} = 4F_{rear} \tag{9.3}$$

For the case when the payload is dropped, the Load taken by the landing gear is given below,

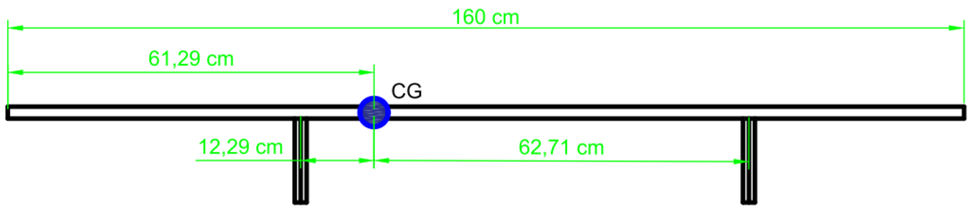


Figure 9.3: Landing gear location without payload

The moment balance equation for this configuration is given as,

$$(12.29)F_{front} = (62.71)F_{rear} \tag{9.4}$$

Thus,

$$F_{front} = 5.1F_{rear} \tag{9.5}$$

In this case without payload the front landing gear component will be taking 83.6 % of the load and the rear landing gear component will take about 16.4 % of the load.

9.4 Design of the landing gear

The landing gear is designed in the software FUSION 360 and also analysed with the impact load plus the weight taken by the landing gear.

9.4.1 Ground clearance required

The ground clearance required is determined as follows, The horizontal propeller on landing should be safe from the damage on hitting the ground. Thus the diameter of the horizontal propellor used is 20 inches (50.8 cm). Thus the radius of the propellor is

25.4 cm. The diameter of the fuselage at maximum diameter is 23 cm. The radius of the fuselage is 13.5 cm . Thus keeping a ground clearance of 3 cm, the height of the landing gear placed at maximum radius must be equal to,

$$H_{FrontLG} = 25.4 \text{ cm} + 3\text{cm} - 13.5 \text{ cm}$$

$$H_{FrontLG} \simeq 15 \text{ cm}$$

9.4.2 Front landing gear component

The front landing gear component is designed to be placed at the maximum diameter and hence the attached image gives the details of the design.

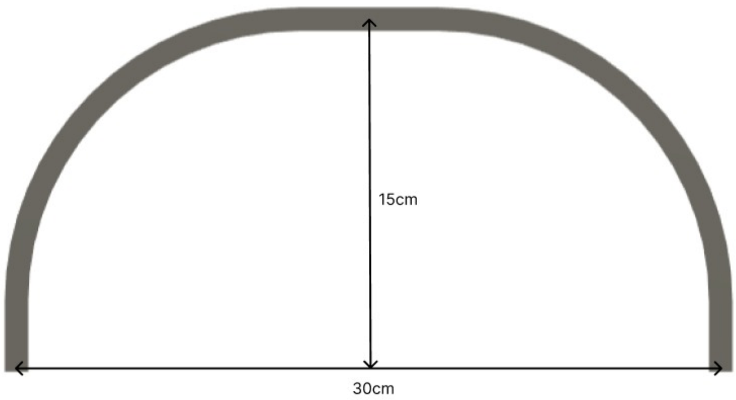


Figure 9.4: Front view of the front landing gear

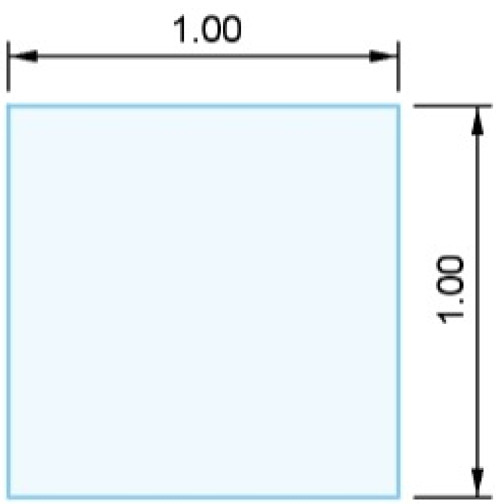


Figure 9.5: Cross section dimensions of the front landing gear

A point to note is that the dimensions marked are in **cm**. Now the material we use is **Aluminium 6061 T6**. The simulation for the loading is to follow, but before that we will measure the load taken by the front landing gear.

$$F_{total} = 225.63 \text{ N} + 93.6 \text{ N} = 319.23 \text{ N}$$

The load is due to the summation of both the impact loading and weight of the VTOL. The front landing gear component takes about 80 % of the maximum load. Thus,

$$F_{total} = (0.8)\,319.23 = 255.3 \text{ N}$$

In the simulations yet to follow this is load applied to the landing gear component. The simulated results are attached.

Figure 9.6: Stress analysis of the front landing gear

The yeild stress taken from internet is more than 100 MPa. Thus our values of the stress are within the elastic limit of the metal.

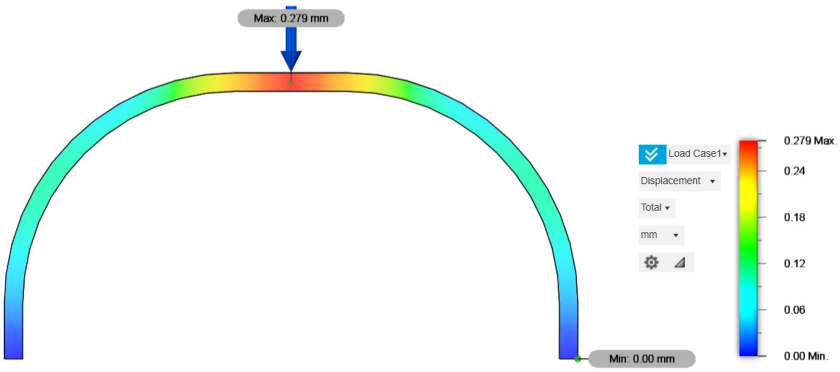


Figure 9.7: Strain/Deformation analysis of the front landing gear

The deformation are less than 1mm and hence the model is very well applicable for the purpose.

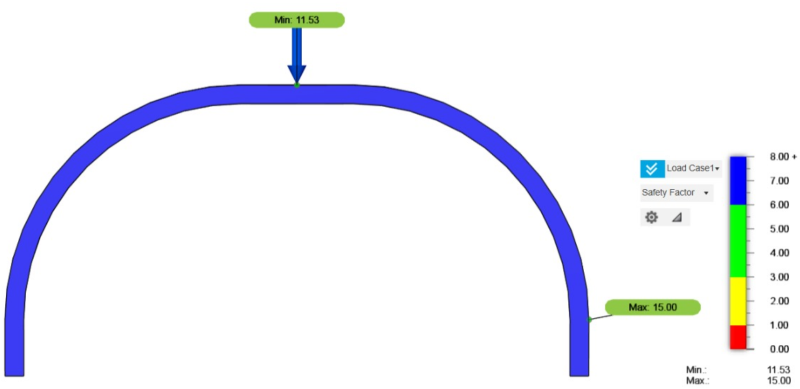


Figure 9.8: Safety factor of the front landing gear

The safety factor found is large meaning to say that the landing gear we use is very safe. From the fusion analysis we got the weight of the landing gear (Front alone) to be **135 gms**.

9.4.3 Rear landing gear component

The back landing gear component is designed to be placed at the a fixed location from the CG of the airplane where the radius of the fuselage is not at maximum diameter of fuselage.

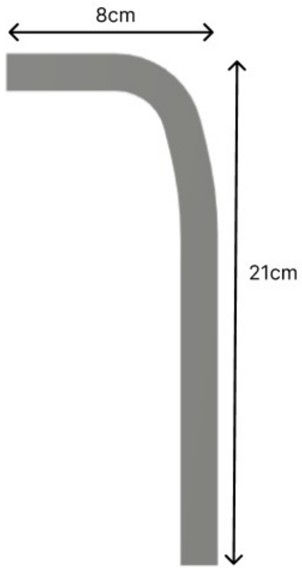


Figure 9.9: Side view of the back landing gear

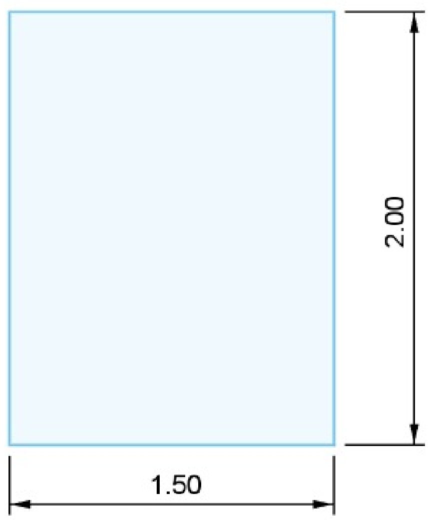


Figure 9.10: Cross section dimensions of the back landing gear

A point to note is that the dimensions marked are in **cm**. Now the material we use is **Aluminium 6061 T6**. The simulation for the loading is to follow, but before that we will measure the load taken by the front landing gear.

$$F_{total} = 225.63 \text{ N} + 93.6 \text{ N} = 319.23 \text{ N}$$

The load is due to the summation of both the impact loading and weight of the VTOL. The back landing gear component takes about 20 % of the maximum load. Thus,

$$F_{total} = (0.2)\,319.23 = 63.846 \text{ N}$$

In the simulations yet to follow this is load applied to the landing gear component. The simulated results are attached.

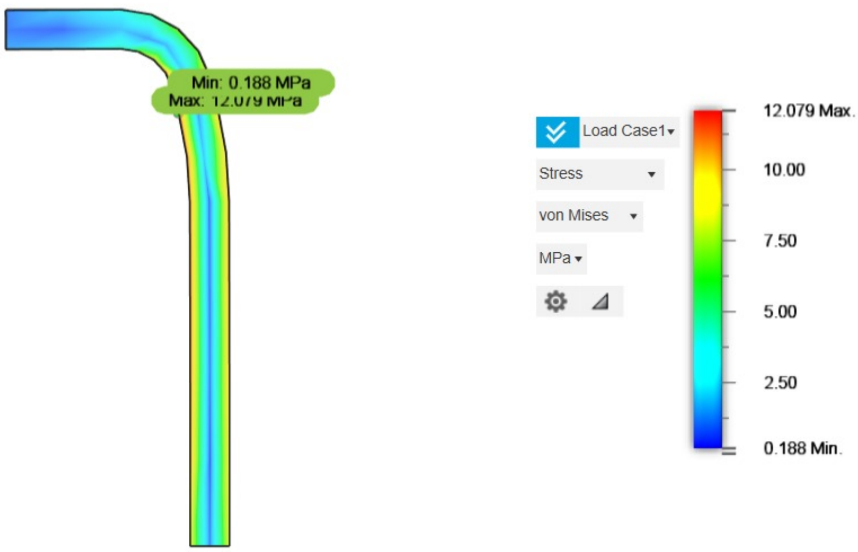


Figure 9.11: Stress analysis of the back landing gear

The yield stress taken from internet is more than 100 MPa. Thus our values of the stress are within the elastic limit of the metal.

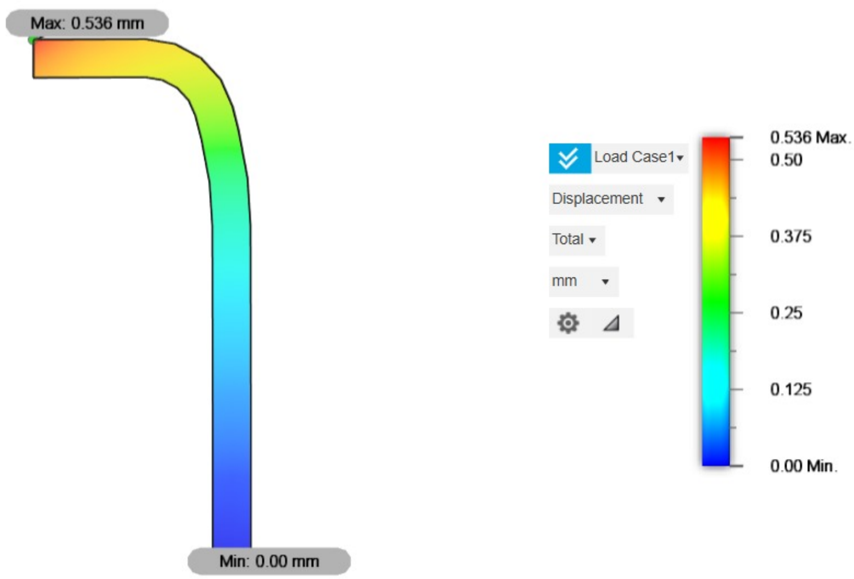


Figure 9.12: Strain/Deformation analysis of the back landing gear

The deformation are less than 1mm and hence the model is very well applicable for the purpose.

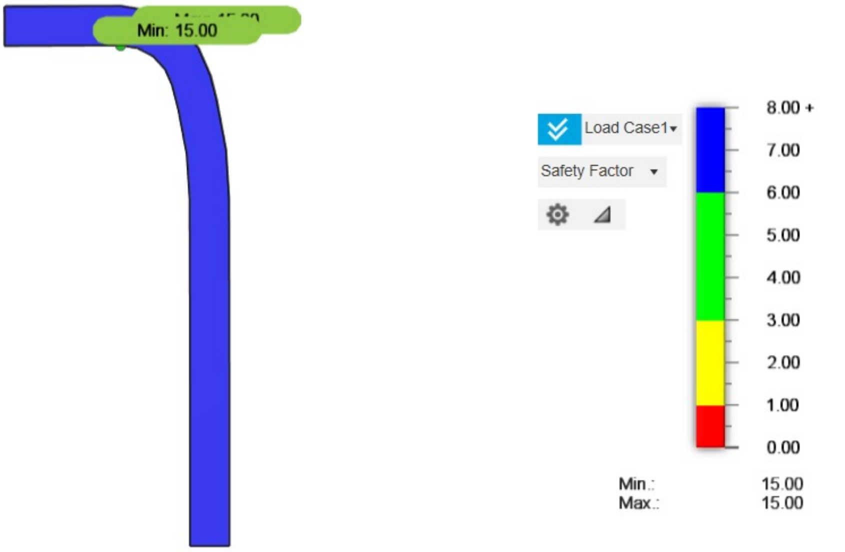


Figure 9.13: Safety factor of the back landing gear

The safety factor found is large meaning to say that the landing gear we use is very safe. From the fusion analysis we got the weight of the landing gear (Back alone) to be **105 gms**.

9.5 Some important checks in landing gear analysis

9.5.1 Entire load on rear landing gear

We have to consider some cases in designing landing gears. First one being that the total is applied to the back landing gear. Now we will run the Fusion360 analysis by applying the entire load on the back landing gear.

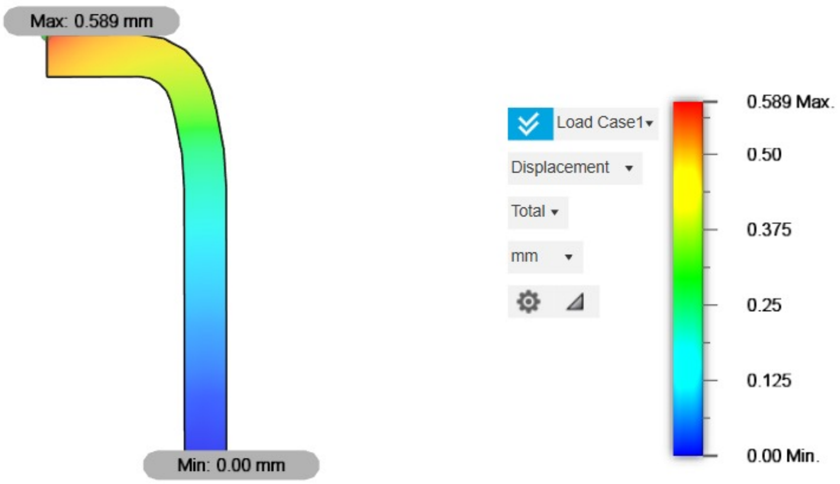


Figure 9.14: Entire impact load on rear landing gear

By applying the entire load we could see that the deflection is less than 0.6 mm . For such small deflection the landing gear is out of danger. Also there is no buckling analysis required for this small deflections as the gears are made up of aluminum which are strong enough.

9.5.2 Entire load on front landing gear

Lets also apply the total load in the front landing gear such that to check the deflections.

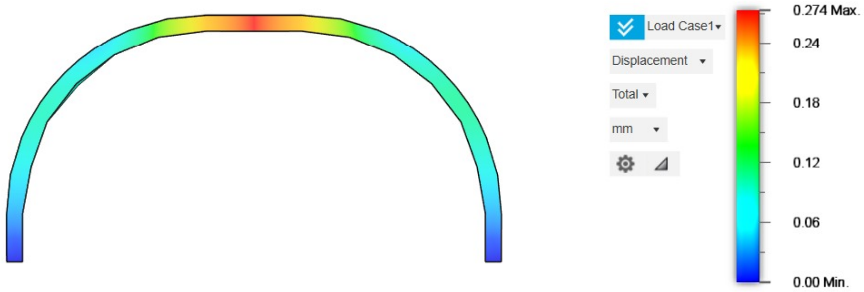


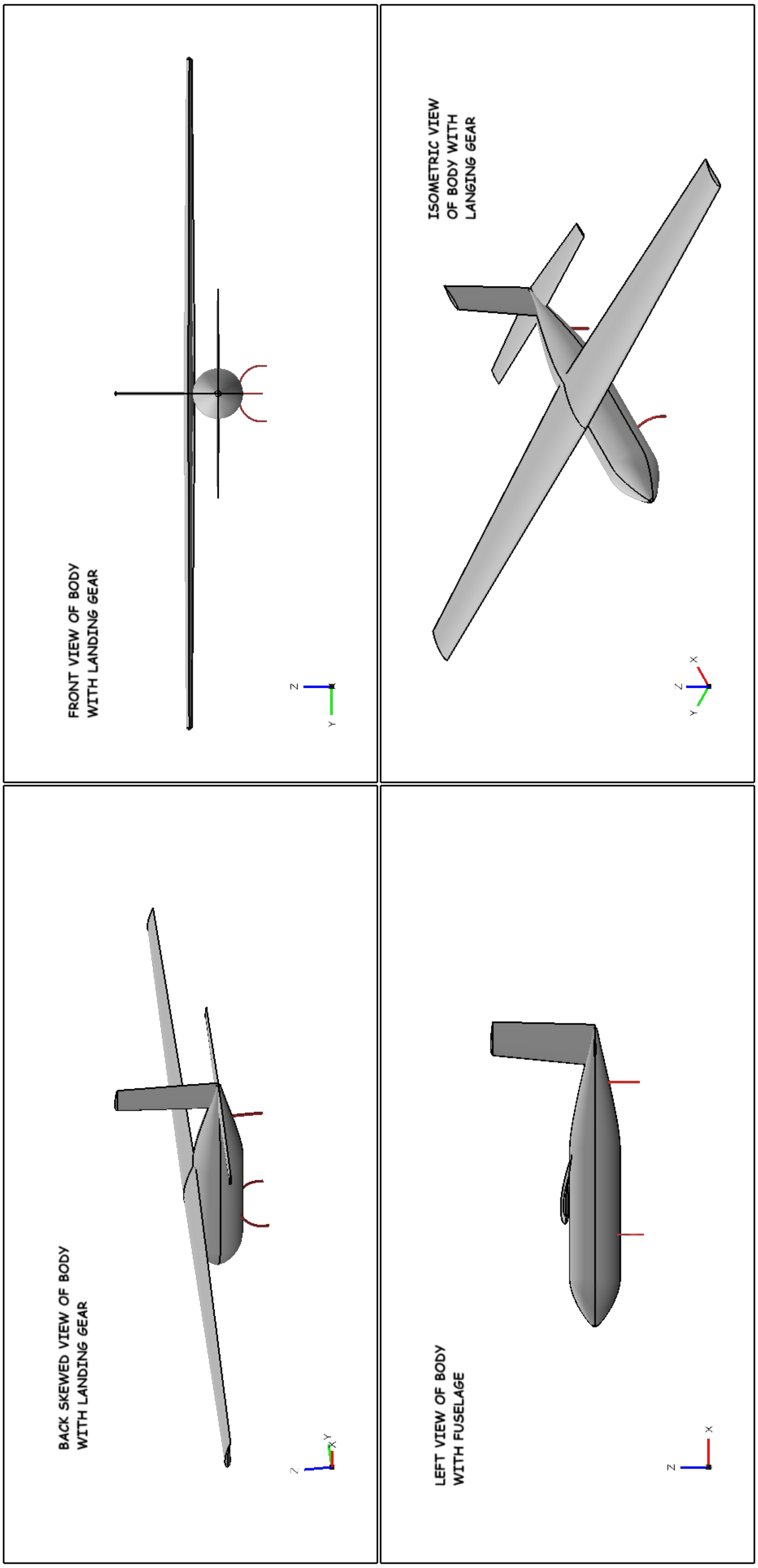
Figure 9.15: Entire impact load on front landing gear

Thus from these analysis we can say that this landing gear will be able to handle the disturbances during the landing process.

9.6 Integrated CAD Model

The first CAD model of the VTOL without power plant but with added landing gear system is attached in the next page. With this the landing gear design and analysis is complete.

Different views of the body of the VTOL without powerplant integrated with landing gear is shown



Chapter 10

Center of Gravity Estimation

In this chapter we will be looking on the aspects of placing the components on the fuselage, calculating the weight of the individual components and calculation of the center of gravity and attachment of the CAD model.

10.1 Internal layout

In this section we will look into the internal components placed in the fuselage. The components that are placed in the fuselage is added in the table attached below,

Component names	Dimensions (x*y*z) (cm)	Placed location (cm)
Control system	10 * 10 * 10	10 cm - 20 cm
Battery	25.6 * 19.3 * 8	25 cm - 48.6 cm
Payload	Cylinder L:53cm Dia:20cm	49 cm - 102 cm
Avionics	20 * 10 * 10	103 cm - 123 cm

Table 10.1: Internal components placed in fuselage

To visualize the components placed in the fuselage the following layout image is attached.

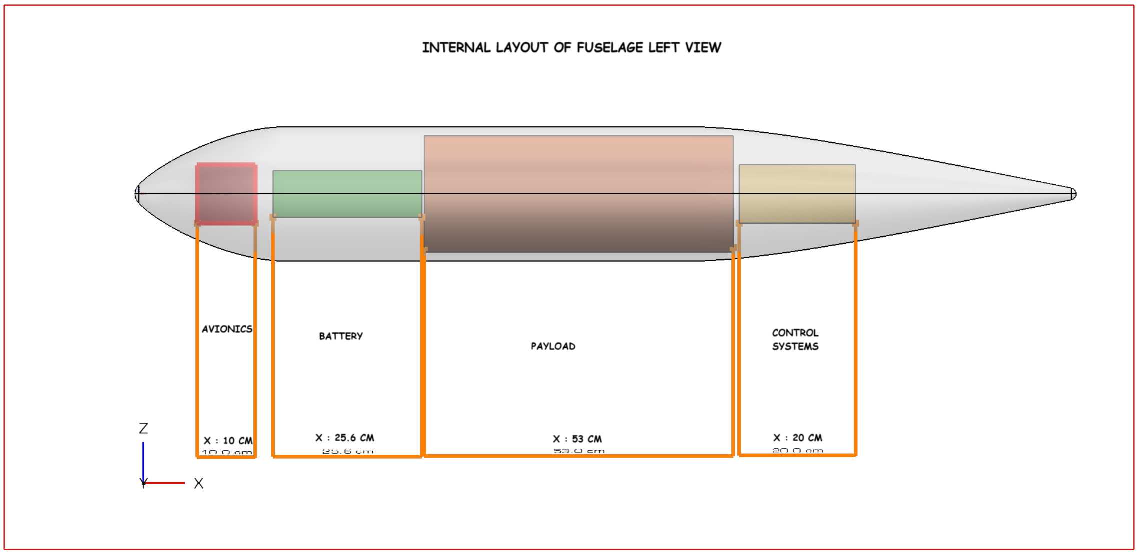


Figure 10.1: Internal layout of fuselage

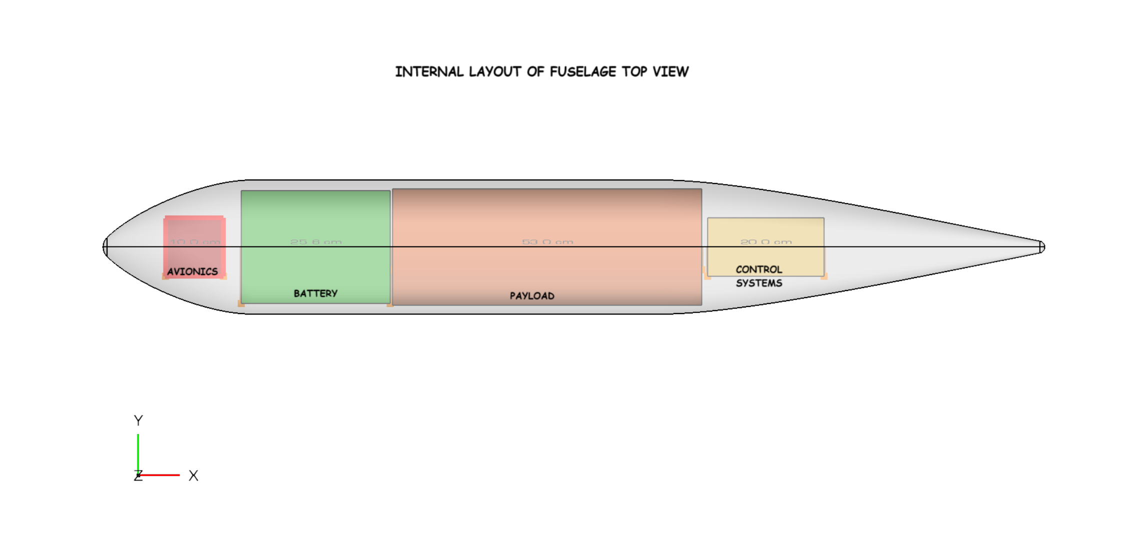


Figure 10.2: Internal layout of fuselage

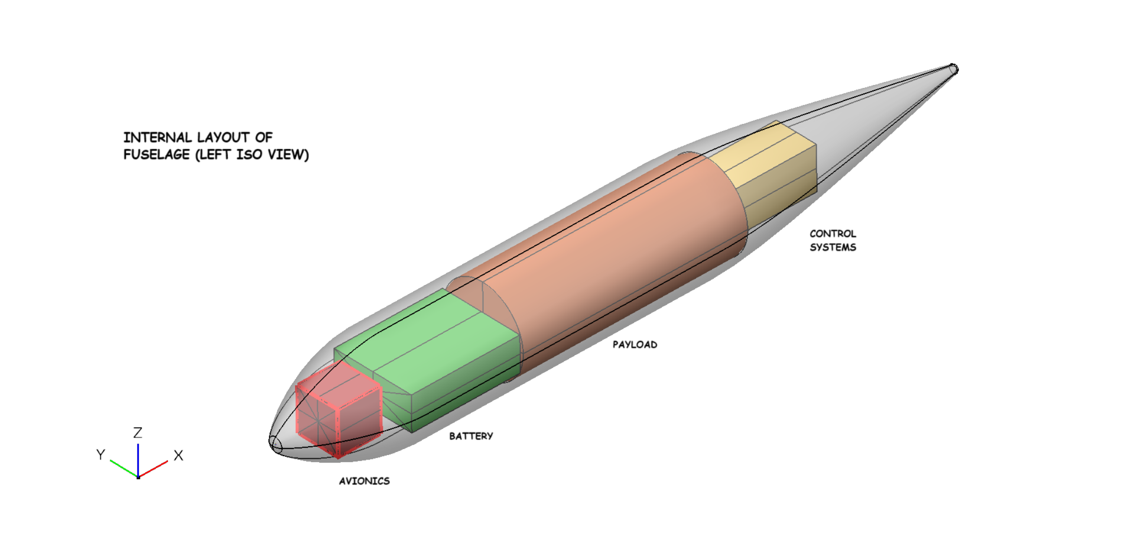


Figure 10.3: Internal layout of fuselage

10.2 Weight of Components

We will be calculating individual weights of various components present in our VTOL, which will be the basis for calculating the empty weight of the VTOL.

10.2.1 Weight of wings

Weight of wing is a function of planform area, maximum thickness to chord ratio, density of material used and various other parameters which is given by below equation from Aircraft Design, M.Sadraey [26] Chapter 10, Equation 10.3:

$$m_W = S_W \cdot MAC \cdot \left(\frac{t}{c}\right)_{max} \cdot \rho_{mat} \cdot K_\rho \cdot \left(\frac{AR \cdot n_{ult}}{cos\Lambda_{0.25}}\right)^{0.6} \cdot \lambda^{0.04} \tag{10.1}$$

Here,

- S_W = Wing planform area = 1.08 m^2

- MAC = Mean Aerodynamic Chord Length = $\frac{2}{3}\frac{1+\lambda+\lambda^2}{1+\lambda}c_r$
- λ = Taper ratio = 0.58
- c_r = Root chord length = 0.38m
- $\Rightarrow$ MAC = 0.30727 m
- $(\frac{t}{c})_{max}$ = Maximum thickness to chord ratio = 0.126 (for wortmann fx 60-126 airfoil)
- ρ_{mat} = Density of the material used.
- K_ρ = Wing density factor = 0.002
- AR = Aspect ratio of wing = 12
- n_{ult} = 1.5 n_{max} = 1.5 $\times$ 2.5 = 3.75
- $\Lambda_{0.25}$ = Quarterchord sweep angle = 0^o

Substituting these values in the equation [10.1]

$$m_W = (1.08)\cdot(0.30727)\cdot(0.126)\cdot\rho_{mat}\cdot(0.002)\cdot\left(\frac{(12)\cdot(3.75)}{cos(0^o)}\right)^{0.6}\cdot(0.58)^{0.04}$$

$$m_W = 8.031\rho_{mat}\times 10^{-4}\text{ kg}$$

for Aerospace grade aluminium, density = 2771 kgm^{-3} $\Rightarrow$ Mass of wing = **2.225 kg**

10.2.2 Weight of horizontal tail

Weight of horizontal tail is a function of planform area, maximum thickness to chord ratio, density of material used and various other parameters which is given by below equation from Aircraft Design , M.Sadraey [26], Chapter 10, Equation 10.5:

$$m_{HT} = S_{HT}\cdot MAC_{HT}\cdot\left(\frac{t}{c}\right)_{max_{HT}}\cdot\rho_{mat}\cdot K_{\rho_{HT}}\cdot\left(\frac{AR_{HT}}{cos\Lambda_{0.25_{HT}}}\right)^{0.6}\cdot\lambda_{HT}^{0.04}\cdot\bar{V}_H^{0.3}\cdot\left(\frac{C_e}{C_t}\right)^{0.4} \quad (10.2)$$

Here,

- S_{HT} = Horizontal tail planform area = 0.172 m^2
- MAC_{HT} = Mean Aerodynamic Chord Length for Horizontal tail = 0.1375 m
- λ_{HT} = Taper ratio = 1
- $(\frac{t}{c})_{max_{HT}}$ = Maximum thickness to chord ratio = 0.12 (for NACA0012 airfoil)
- ρ_{mat} = Density of the material used
- $K_{\rho_{HT}}$ = Horizontal tail density factor = 0.0175
- AR = Aspect ratio of horizontal tail = 8
- $\Lambda_{0.25}$ = Quarterchord sweep angle = 0^o
- $\bar{V}_H$ = Horizontal tail Volume Ratio = $\frac{S_{HT}\cdot lh}{S_W\cdot b}$
- lh = Tail arm = 0.96 m
- b = Wing average chord = Wing area/Wing span = 1.08/3.6 = 0.3 m
- $\Rightarrow$ $\bar{V}_H$ = 0.50963

- C_e = Elevator chord length = 0.054 m
- C_t = Tail chord length = MAC_{HT} = 0.1375 m
- $\Rightarrow \frac{C_e}{C_t}$ = 0.39273
-

Substituting these values in the equation [10.2]

$$m_{HT} = (0.172) \cdot (0.1375) \cdot (0.12) \cdot \rho_{mat} \cdot (0.0175) \cdot \left(\frac{8}{cos(0^o)}\right)^{0.6} \cdot (0.50963)^{0.3} \cdot (0.39273)^{0.4} \quad (10.3)$$

$$m_{HT} = 9.72\rho_{mat} \times 10^{-5} \text{ kg}$$

for Aerospace grade aluminium, density = 2771 kgm^{-3} $\Rightarrow$ Mass of horizontal tail = **0.2635 kg**

10.2.3 Weight of vertical tail

Weight of vertical tail is a function of planform area, maximum thickness to chord ratio, density of material used and various other parameters which is given by below equation from Aircraft Design , M.Sadraey [26], Chapter 10, Equation 10.6:

$$m_{VT} = S_{VT} \cdot MAC_{VT} \cdot \left(\frac{t}{c}\right)_{max_{VT}} \cdot \rho_{mat} \cdot K_{\rho_{VT}} \cdot \left(\frac{AR_{VT}}{cos\Lambda_{0.25_{VT}}}\right)^{0.6} \cdot \lambda_{VT}^{0.04} \cdot \bar{V}_V^{0.2} \cdot \left(\frac{C_r}{C_v}\right)^{0.4} \quad (10.4)$$

Here,

- S_{HVT} = Vertical tail planform area = 0.1215 m^2
- MAC_{VT} = Mean Aerodynamic Chord Length for Vertical tail = 0.2025 m
- λ_{VT} = Taper ratio = 1
- $(\frac{t}{c})_{max_{VT}}$ = Maximum thickness to chord ratio = 0.12 (for NACA0012 airfoil)
- ρ_{mat} = Density of the material used
- $K_{\rho_{VT}}$ = Vertical tail density factor = 0.052
- AR = Aspect ratio of vertical tail = 2.96
- $\Lambda_{0.25}$ = Quarterchord sweep angle = 0^o
- $\bar{V}_v$ = Vertical tail Volume Ratio = $\frac{S_{VT} \cdot lh}{S_W \cdot b}$
- lh = Tail arm = 0.96 m.
- b = Wing average chord = Wing area/Wing span = 1.08/3.6 = 0.3 m
- $\Rightarrow \bar{V}_v$ = 0.36
- C_r = Rudder chord length = 0.0643 m
- C_v = Tail chord length = MAC_{VT} = 0.2025 m
- $\Rightarrow \frac{C_r}{C_v}$ = 0.31753

Substituting these values in the equation [10.3]

$$m_{HT} = (0.1215) \cdot (0.2025) \cdot (0.12) \cdot \rho_{mat} \cdot (0.052) \cdot \left(\frac{2.96}{cos(0^o)}\right)^{0.6} \cdot (1)^{0.04} \cdot (0.36)^{0.2} \cdot (0.31753)^{0.4}$$

$$m_{VT} = 1.5066\rho_{mat} \times 10^{-4} \text{ kg}$$

for Aluminium, density = 2711 kgm^{-3} $\Rightarrow$ Mass of vertical tail = **0.4087 kg**

10.2.4 Weight of fuselage

Weight of fuselage is a function of density of material used and various other parameters which is given by below equation from Aircraft Design , A systems engineering approach book, Chapter 10, Equation 10.7:

$$m_F = L_f \cdot D^2_{f_{max}} \cdot \rho_{mat} \cdot K_{\rho_f} \cdot n^{0.25}_{ult} \cdot K_{inlet} \tag{10.5}$$

Here,

- L_f = Fuselage length = 1.6 m
- $D_{f_{max}}$ = Fuselage maximum diameter = 0.29 m
- ρ_{mat} = Density of the material used
- K_{ρ_f} = Fuselage density factor = 0.003
- n_{ult} = 3.75 (from section 10.1.1)
- K_{inlet} = 1 as inlets are not present on the fuselage

Substituting these values in the equation [10.4]

$$m_F = (1.6) \cdot (0.29)^2 \cdot \rho_{mat} \cdot (0.003) \cdot (3.75)^{0.25} \cdot (1)$$

$$m_F = 5.99202\rho_{mat} \times 10^{-4} \text{ kg}$$

for Aluminium 6061, density = 2711 kgm^{-3} $\Rightarrow$ Mass of fuselage = **1.624 kg**.

10.3 Calculation of Center of gravity

In this sections from the weights and the placement of the components now we are in a position to calculate the Center of gravity of the aircraft. The components and their placements are given in the table below,

Component	Weight (kg)	Location of COM (x) [cm]	Location of COM (z) [cm]
Battery	8	35.8	-4.6126
Payload	4	74.8	0
Control System	1	15	0
Avionics	1	113	0
Wing	2.225	64	14
Vertical tail	0.4087	150.36	22.587
Horizontal tail	0.2635	146.5	0
Landing gear	0.24	81.8125	-14.5
Fuselage	1.624	72	0
Horizontal powerplant	0.2	72	0
Vertical powerplant	3.5	58	0

Table 10.2: Components and their placements

A point to note is that the battery is adjusted in the vertical direction such that the center of gravity of the plane is along the line connecting the nose and the tail. The center of gravity for the fuselage and vertical tail are calculated using the OpenVSP software. The wing is placed at 40 % of the fuselage length from the nose, as in the 8th chapter we calculated the tail arm as 60 % of the fuselage length as given in [26].

The CG calulation is as follows,

$$CG = \frac{\Sigma W_i x_i}{\Sigma x_i} \tag{10.6}$$

From the values in the above table and equation [10.6], the center of gravity is calculated to be at a location of **CG = 58 cm** from the nose, h_{CG} **= 0.3625 cm**. Note that in the calculation of the CG we did not include the vertical powerplant as the it will placed such that the CG is unaltered. We will see this in the next section.

10.4 Placement of propellers and motors

After finding the center of gravity of the plane without power plant we design the placement of the motor/propeller combination such that the CG is not affected. The placement of the horizontal propeller is at the nose. From the research paper and reference from the internet [37], it is found that optimal placement of propeller is 1.5 times the diameter. Thus the frame is a square of side **a = 84 cm**. The CAD model is attached below,

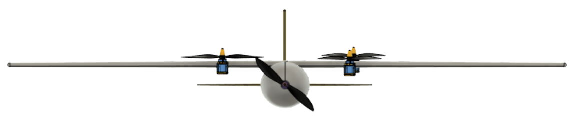


Figure 10.4: Front view

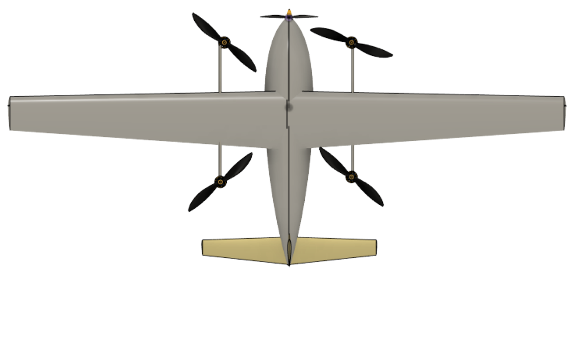


Figure 10.5: Top view

Figure 10.6: Side view

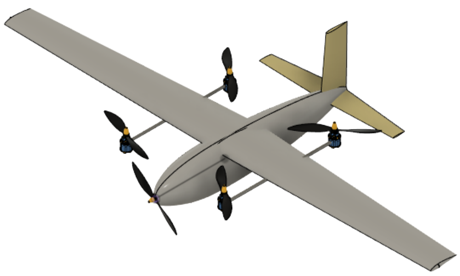


Figure 10.7: Isometric view

Chapter 11

Analysis of stability

In this chapter we will be looking into the stability analysis of our VTOL. We will also estimate the parameters left out in the previous chapter.

11.1 Analysis of Trim

In this section we will look the trim condition of the VTOL at cruise condition into the parameters already found that are helpful in the trim equation.

Aerodynamic Parameters	**Values**
C_{L0}^{wb}	0.3465
$C_{L_\alpha}^{wb}$	0.09258 /degrees
Wing incidence	5.5 degrees
MAC of wing ($\bar{c}$)	0.3 m
$C_{m_{AC}}^{wb}$	-0.12539
$C_{D_{AC}}^{wb}$	0.03
$Z_{AC}^{w}/\bar{c}$	0.5
X_{AC}^{wb}	0.596 m
X_{CG}	0.58 m
$h_{AC}^{wb} - h_{CG}$	0.0533
HTVR (V_H)	0.5
$C_{L_\alpha}^{t}$	0.10 /degree

Table 11.1: Parameters required for stability analysis

The above parameters (Here wb stands for wing-body) , in the table were found from previous chapters and some of them from the XFLR5 software simulation, in which some of them are attached below.

11.1.1 Parameters required

In this subsection we will look into finding the parameters in the table attached above. For the aerodynamic coefficients we used the plots from XFLR5 to calculate the values for the wing. Refer fig.6.17. Some of the important dimensions of our VTOL are attached below.

From the below diagram, we will find the parameters involving the dimension of the Vtol.

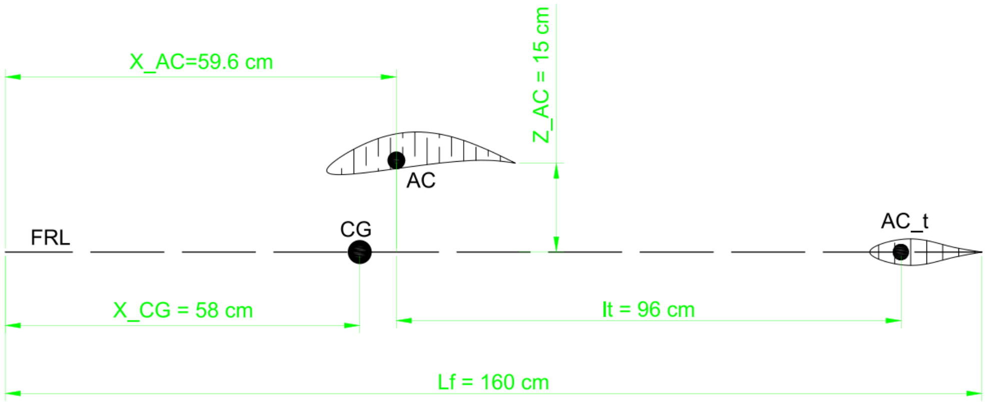


Figure 11.1: Dimensions in cm

For the location of wing aerodynamic center and the moment coefficient at the aerodynamic center, we used XFLR5 software, by changing the moment reference location and checking whether the moment remains constant at different angles of attack.

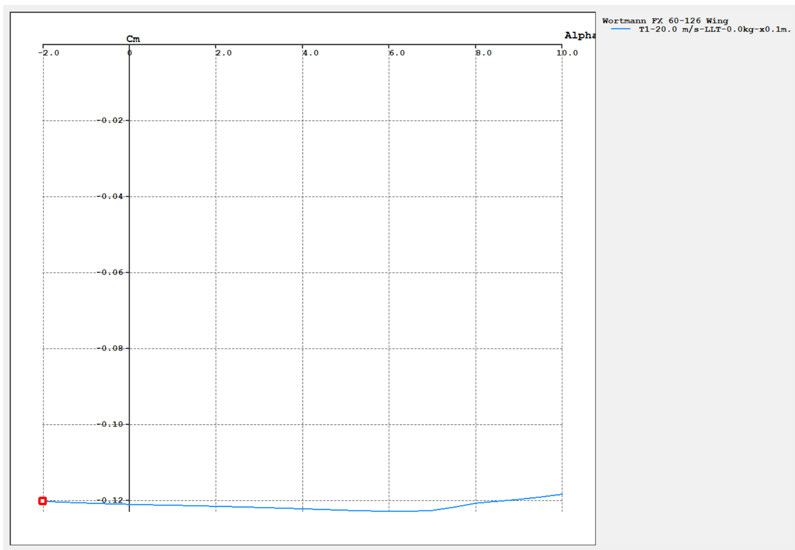


Figure 11.2: $C_{m_{AC}}^{wb}$ VS α for wing

From the above diagram and moment reference location the distance of the aerodynamic center from the leading edge is **9.2 cm** and the corresponding percentage of MAC is **30%**. Thus at this location the $C_{m_{AC}}^{wb}$ was found to be **-0.12539**.

The tail (horizontal) aerodynamic coefficient is found from XFLR5 with dimensions from the previous chapters.

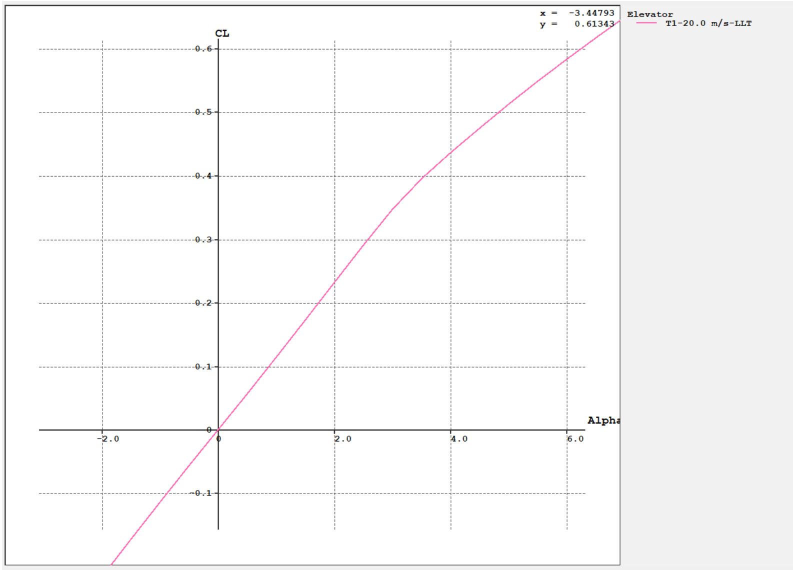


Figure 11.3: Horizontal tail C_l

11.1.2 Trim condition

From Chapter 3 of N.K.Sinha [38],for our configuration the pitching moment at the Cg was found out to be,

$$C_{m_{CG}} = C^{wb}_{m_{AC}} - C^{wb}_{L}(h^{wb}_{AC} - h_{cg}) + C^{wb}_{D}\left(\frac{Z^{wb}_{AC}}{\bar{c}}\right) - \eta V_H C^{t}_{L} cos(\epsilon) + \eta V_H C^{t}_{D} sin(\epsilon) \quad (11.1)$$

Thus the $C_{m\alpha}$ per degree is negative for the chosen parameters. The last term in the equation is neglected as the downwash angle (ϵ) is small. Also, note that the tail is placed along the FRL (Fuselage reference line) and the wing is kept at an incidence angle of 5.5 degrees such that at cruise the FRL line is horizontal.

By substituting the values in the eq.11.1 from the table.11.1 taking $\eta = 0.9$ is the efficiency factor of the dynamic pressure at the tail to the dynamic pressure at the wing. Here the downwash is small and so there is not much change in velocity.
By taking the value of $C_{m_{CG}}$=0 required for trim at cruise, and substituting the values of the parameters known we get,

$$\boxed{{C_L}^t = -0.300}$$

Thus for this value, now we are in a position to find the tail incidence angle required to keep the plane in trimmed condition on a level horizontal cruise. From Chapter 3 of N.K.Sinha [38],eq(3.13) and eq(3.5),

$$\epsilon = \frac{2C^{wb}_{L}}{\pi(AR_w)} \quad (11.2)$$

Where C^{wb}_L = 0.85 is the cruise lift from wing, AR_w = 12 is the aspect ratio of the wing. From this we get the downwash angle to be around,

$$\boxed{\epsilon = 2.5862^\circ}$$

From another equation, we have,

$${C_L}^t = C_{L_{\alpha^t}}(\alpha - i_t - \epsilon) \quad (11.3)$$

Where i_t is the tail incidence angle and α is the angle of attack. Here α = 0 in cruise as the plane is horizontal and then from the values from table.11.1 we get,

$$-2.57 = (0 - i_t - 2.58) \quad (11.4)$$

$$\boxed{i_t = -0.01^\circ}$$

11.2 Longitudinal Stability

11.2.1 Neutral point

It is the aerodynamic center of the plane, the moment about this point is independent of angle of attack. From Chapter 3 of N.K.Sinha [38], eqn(3.27) for our configuration of wing,

$$h_{NP} = h_{AC}^{wb} + \eta V_H \frac{C_{L_\alpha}^t}{C_{L_\alpha}^{wb}}(1 - \epsilon_\alpha) \tag{11.5}$$

Where ϵ_α is ,

$$\frac{\partial \epsilon}{\partial \alpha} = \frac{2C_{L\alpha_w}}{\pi AR} = 0.0049 \tag{11.6}$$

Note that all values are calculated per degree. Substituting these values in the equation 11.4 we get,

$$h_{NP} = h_{AC}^{wb} + 0.4836 \tag{11.7}$$

$$h_{NP} = 1.986 + 0.4836 \tag{11.8}$$

Thus h_{NP} = 2.469. The value of X_{NP} = 74.08 cm. **The distance of the neutral point from the nose is 71.23 cm.**

11.2.2 Static Margin

For a stable UAV the neutral point should be behind the CG location, therefore the static margin should be positive. The non dimensional distance between the CG and the NP is the static margin.

$$S.M = h_{NP} - h_{CG} \tag{11.9}$$

$$\boxed{S.M = 0.4836} \tag{11.10}$$

11.2.3 Moment Coefficients

The value of the moment at our trim condition is zero, the trim angle is zero (i.e. the plane is horizontal, parallel to velocity). Thus the value of C_{m_0} = 0. And the value of C_{m_α} has to be negative to have pitch stability.

$$C_m = C_{mo} + C_{m\alpha}\alpha \tag{11.11}$$

For our configuration of the wing and tail we have plotted the moment and lift coefficient using XFLR5 is attached below,

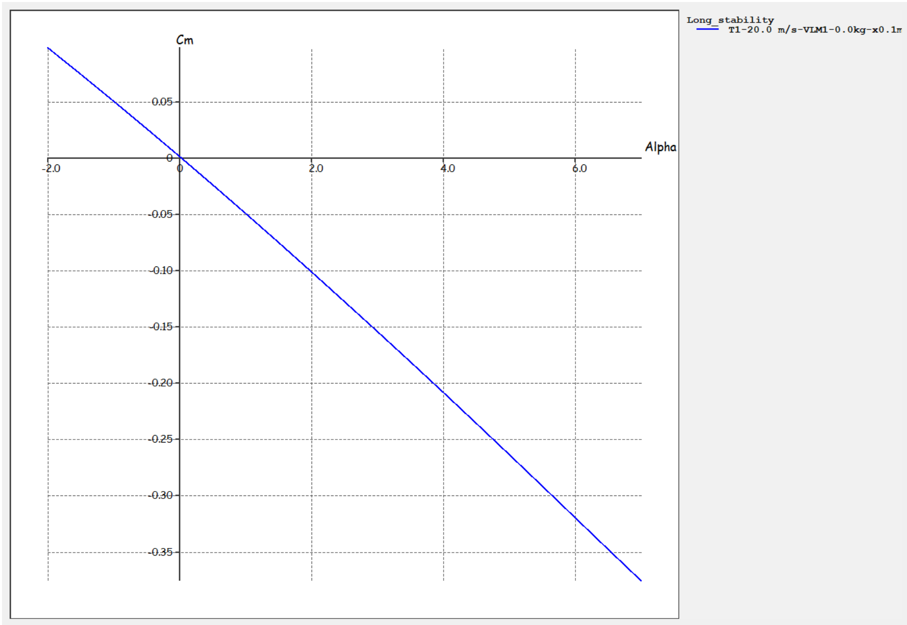


Figure 11.4: Moment coefficient for wing tail configuration

To check the sign for $C_{m\alpha}$:

$$C_{m_\alpha} = C_{L_\alpha}^{wb}(h_{CG} - h_{AC}^{wb}) + C_{D_\alpha}\left(\frac{Z_{AC}^{wb}}{\bar{c}}\right) - C_{L_\alpha}^{t} V_H(1 - \epsilon_\alpha) \tag{11.12}$$

$$\boxed{C_{m\alpha} = -0.03926} \tag{11.13}$$

Thus the $C_{m\alpha}$ per degree is negative for the chosen parameters.

11.3 Lateral Stability

In lateral stability analysis, we will be looking into the coupled analysis of yaw and roll stability. The directional and roll stability derivatives are computed as follows: The equations were taken from Raymer [31]. The values used for computing the following stability derivatives are mentioned below:

Parameters	Values
eta_v	0.9
l_{vt}	0.96m
k_{fl}	0.75
Volume	$3.545 \times 10^{-4} m^3$
D_f	0.27m
W_f	0.27m
$C_{L\alpha_h}$	0.1/deg
τ_e	0.55

Table 11.2: Parameters used for evaluating stability derivatives

The τ_v value is determined from this graph in Fig 11.5

11.3.1 Directional Stability

This is concerned with the aircraft's ability to yaw or weathercock into the wing to maintain stability, thus a positive sideslip should account for a positive yawing moment.

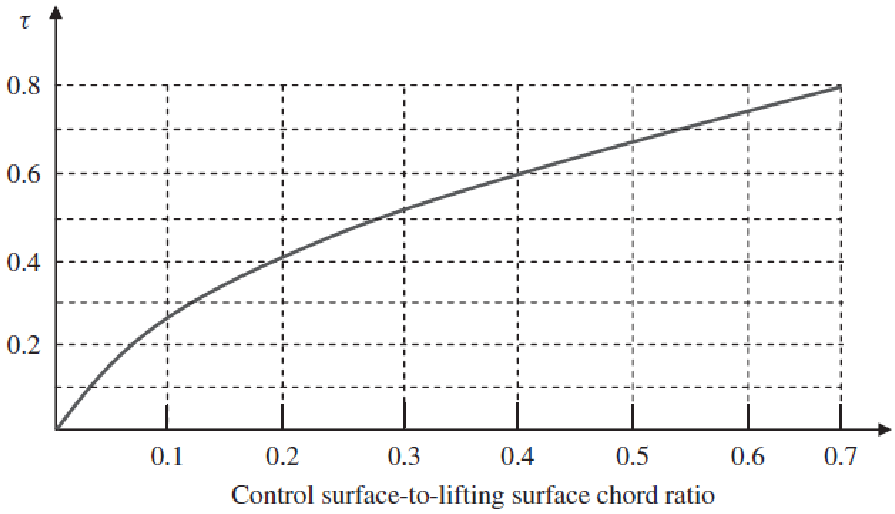


Figure 11.5: Control Surface angle of attack lifting parameter [26]

The value of Cn_β for a statically directional stable aircraft is positive. The contribution for $C_{n\beta}$ is from wing, tail and fuselage.

$$C_{n\beta} = C_{n\beta w} + C_{n\beta fus} + C_{n\beta v} \tag{11.14}$$

$$\frac{\partial Cn}{\partial \beta_v} = Cl_{v\alpha}\eta_v \frac{l_{vt}S_v}{bS} K_{fl} \tag{11.15}$$

Here the K_{fl} is the correction factor with values ranging from 0.65 to 0.85, for our purpose we have taken an average value of 0.75

$$\boxed{Cn_{\beta v} = 0.0275} \tag{11.16}$$

$$Cn_{\beta w} = C_L^2(\frac{1}{4\pi A})(1 - A/2 - A^2/8) \tag{11.17}$$

$$\boxed{Cn\beta w = -0.0097} \tag{11.18}$$

$$Cn\beta fus = -1.3\frac{Volume}{S_w b}(\frac{D_f}{W_f} \tag{11.19}$$

$$\boxed{Cn\beta fus = -0.000118} \tag{11.20}$$

On combining the above equations, Cn_β comes out as positive. As $Cn_\beta > 0$, Thus the UAV has static directional stability.

11.3.2 Roll Stability

Roll stability reflects the ability of aircraft to counter any imbalance in lift-causing roll, thus for a positive sideslip, the roll moment should be negative. For static stability in roll the $\frac{\partial Cl}{\partial \beta}$ should be negative. Considering the contribution from the wing and vertical tail, we have the following equations:

$$Cl_\beta = Cl_{\beta w} + Cl_{\beta v} \tag{11.21}$$

Contribution in roll coefficient due to wing:

$$Cl_{\beta w} = (\frac{Cl_{\beta wing}}{C_L})C_L + (Cl_\beta)_\Gamma) + Cl_{\beta wf} \tag{11.22}$$

Contribution in roll coefficient due to dihedral angle in the wing:

$$(Cl_\beta)_\Gamma) = -\frac{C_{L\alpha}\Gamma}{4}[\frac{2(1+2\Gamma}{3(1+\Gamma}] \tag{11.23}$$

In our case since there is no dihedral, $(Cl_\beta)_\Gamma)$ goes to zero.

$$\boxed{(Cl_\beta)_\Gamma) = 0} \tag{11.24}$$

$$Cl_{\beta wf} = -1.2\frac{\sqrt{A}Z_{wf}(D_f + W_f)}{b^2} \tag{11.25}$$

$$\boxed{Cl_{\beta wf} = -0.0259} \tag{11.26}$$

Contribution due to vertical tail.

$$Cl_{\beta v} = -C_{F_{\beta v}}\frac{\partial\beta_v}{\partial\beta}\eta_v\frac{S_v}{S_w}Z_v \tag{11.27}$$

$$\frac{\partial\beta_v}{\partial\beta}\eta_v = 0.724 + \frac{3.06\frac{S_{vs}}{S_w}}{2} - 0.4\frac{Z_{wf}}{D_f} + 0.009A_{wing} \tag{11.28}$$

$$Cl_{\beta v} = -0.00016 \tag{11.29}$$

On combining the contribution due to above, the final $Cl_{\beta v}$ is

$$\boxed{Cl_{\beta v} = -0.02576} \tag{11.30}$$

Thus the final roll stability derivative is negative.

After evaluating our longitudinal and lateral stability parameters, we can proceed to calculate our control stability derivatives.

11.4 Stability derivatives of control surface

In this section, the control derivatives of our VTOL are estimated. The process and analysis of this is as follows: The equations were taken from M.Sadraey [26]

11.4.1 Elevator Design

Considering the values estimated from tail layout design:

Elevator Planform Area	$0.0516m^2$
Elevator chord	0.05m
Elevator span	1m
Elevator Maximum Deflection	20.84 deg

Table 11.3: Elevator Design Parameters

11.4.2 Elevator Stability derivatives

Aircraft pitching moment Coefficient

This derivative shows the UAV's response to the pitching moment due to elevator deflection.

$$Cm_{\delta E} = \frac{\partial C_m}{\partial \delta_E} = -C_{L\alpha h}\eta_h V_H \frac{b_E}{b_h}\tau_e \tag{11.31}$$

$$\boxed{Cm_{\delta_E} = -0.02267/deg} \tag{11.32}$$

Maximum Elevator Deflection angle

The maximum angle is limited by the angle which corresponds to the pitch-up moment which causes the aircraft to stall. Since we have the $C_{m\alpha}$ and Cm_{δ_E} we can compute the maximum deflection angle.

$$C_{m\alpha} = -0.0392 \tag{11.33}$$

It is calculated earlier that the wing stall is 10 degrees after accounting the wing incidence angle. So the moment coefficient when angle of attack is 10 degrees is :

$$C_m = -0.0392 \tag{11.34}$$

Equating this to the pitch-up moment due to elevator deflection we get:

$$C_m = -0.02267\delta_E \tag{11.35}$$

$$\delta_E = -0.0392/ - 0.02267 = 20.8^\circ \tag{11.36}$$

$$\boxed{\delta_{Emax} = 20.8^\circ} \tag{11.37}$$

Aircraft lift coefficient w.r.t elevator deflection

This derivative depicts the aircraft's lift coefficient for an elevator deflection.

$$C_{L\delta_E} = C_{L\alpha_h}\eta_h \frac{S_h}{S}\frac{b_E}{b_h}\tau_e \tag{11.38}$$

$$\boxed{C_{L\delta_E} = 0.00709/deg} \tag{11.39}$$

Tail lift coefficient w.r.t elevator deflection

This gives the tail lift coefficient of UAV for a particular elevator deflection.

$$C_{L_{h_{\delta E}}} = C_{L_{\alpha_h}}\tau_e \tag{11.40}$$

$$\boxed{C_{L_{h_{\delta E}}} = 0.055/deg} \tag{11.41}$$

11.4.3 Rudder Stability derivatives

11.4.4 Rudder Design

Considering the values estimated from the tail layout design:

Rudder Planform Area	$0.018m^2$
Rudder chord	0.06m
Rudder span	0.3m

Table 11.4: Rudder Design Parameters

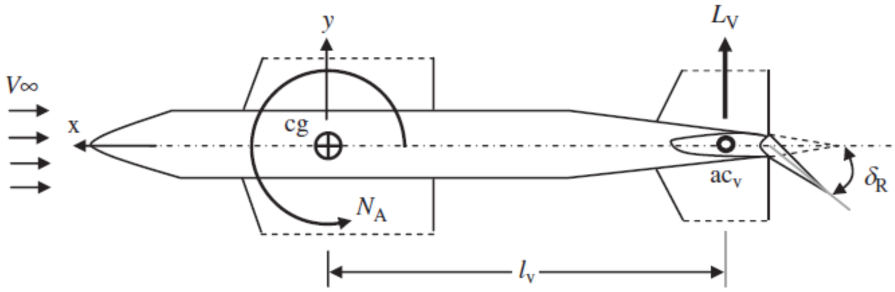


Figure 11.6: Directional control via rudder (top view)

Directional control derivative

This derivative gives the response of the yawing moment given some rudder deflection. Also given some positive rudder deflection, the aircraft will yaw anti-clockwise as viewed from the top. Thus $Cn_{\delta R}$ derivative is negative.

$$Cn_{\delta R} = -C_{L\alpha v} V_v \eta_v \tau_r \frac{b_R}{b_v} \tag{11.42}$$

$$\boxed{Cn_{\delta R} = -0.009914/deg} \tag{11.43}$$

Chapter 12

Performance Analysis

In this chapter, we will be doing the performance analysis of our UAV.

12.1 Estimating Zero Lift Drag (Parasitic drag) (C_{D_0})

The parasitic drag is estimated using Component Buildup method. In this method, flat plate skin-friction coefficient (C_f), Form-Factor (FF), Interference factor (Q) are calculated for each component of the UAV.Here, FF accounts for pressure drag for each component. The total component drag is calculated as the product of ratio of wetted area to reference area of component, C_f, FF and Q. And, Total parasitic drag is estimated as below.

$$C_{Do} = \Sigma_i \left(C_{fi} Q_i FF_i \frac{S_w}{S_{ref}} \right) + C_{D_p} + C_{D_l} \tag{12.1}$$

Here, C_{D_p} and C_{D_l} are the zero lift drag components of vertical propellers and landing gear respectively.

To calculate the friction coefficient, We will use the formula from Raymer [31] Chapter 12 (eq 12.27) which is used to estimate C_f for a flat plate in turbulent conditions given below,

$$C_f = \frac{0.455}{(\log R_e)^{2.58}(1 + 0.144M^2)^{0.65}} \tag{12.2}$$

from the previous chapters we have our values of M = 0.0583 and Reynolds number (R_e) changes according to component as the fuselage has a different characteristic dimension than that of the wing.

12.1.1 Wing

For a wing, the characteristic dimension is its Mean Aerodynamic Chord Length (MAC) = 0.3073 m, velocity (v) = 20 m/s, density of air (ρ) = 1.225 kg/m^3 and viscosity coefficient of air (μ) = 1.825 $\times 10^{-5}$ Pa.s. Hence, corresponding Reynold's numbers is

$R_e = \frac{Lv\rho}{\mu}$ = 412539.7

Substituting values of R_e & M in equation [12.2],

$$C_f = 0.0053$$

Form factor for a wing is calculated as

$$FF_w = \left(1 + \frac{0.06}{(x/c)} \frac{t}{c} + 100(\frac{t}{c})^4 \right) \left[1.34M^{0.18}(cos(\lambda)^{0.28}) \right] \tag{12.3}$$

From section 6.1.2, We know that the airfoil used in the wing is Wortmann FX 60-126. For this airfoil, x/c = 0.279, t/c = 0.126. Sweep angle at maximum thickness line is λ, if is calculated by

$$tan\lambda_n = tan\lambda_m - \frac{4}{AR}\left[(n-m)\frac{1-TR}{1+TR}\right] \tag{12.4}$$

Here, $\lambda_{1/4} = 0^o$, AR = 12, TR = Taper ratio = 0.58, n = 0. Substituting these values in equation [12.4], $\lambda = 1.269^o$. Substituting values of λ, M, x/c and t/c in equation [12.3]

$$FF_w = 1.0413$$

From section 3.2, $\frac{S_w}{S_{ref}} = 2.003$
For High-wing configuration, Q = 1
Substituting these values in equation [12.1]

$$C_{D_w} = 0.01105$$

12.1.2 Horizontal Tail

For horizontal tail, the characteristic dimension is its Mean Aerodynamic Chord Length (MAC) = 0.1375 m, velocity (v) = 20 m/s, density of air (ρ) = 1.225 kg/m^3 and viscosity coefficient of air (μ) = 1.825 $\times 10^{-5}$ Pa.s. Hence, corresponding Reynold's numbers is

$$R_e = \frac{Lv\rho}{\mu} = 184589.04$$

Substituting values of R_e & M in equation [12.2],

$$C_f = 0.0063$$

Form factor for horizontal tail is calculated from equation [12.3].From section 8.6, We know that the airfoil used in the wing is NACA0012. For this airfoil, x/c = 0.3, t/c = 0.12. Sweep angle at maximum thickness line is λ, it is calculated by equation [12.4]. Here, $\lambda_{1/4} = 0^o$, AR = 8, TR = Taper ratio = 1, n = 0. Hence, $\lambda = 0^o$. Substituting values of λ, M, x/c and t/c in equation [12.3]

$$FF_{HT} = 1.0129$$

From section 3.2, $\frac{S_{HT}}{S_{ref}} = 2.003(0.172/1.08) = 0.319$
For Horizontal tail, Q = 1.08
Substituting these values in equation [12.1]

$$C_{D_{HT}} = 0.0022$$

12.1.3 Vertical Tail

For vertical tail, the characteristic dimension is its Mean Aerodynamic Chord Length (MAC) = 0.2025 m, velocity (v) = 20 m/s, density of air (ρ) = 1.225 kg/m^3 and viscosity coefficient of air (μ) = 1.825 $\times 10^{-5}$ Pa.s. Hence, corresponding Reynold's numbers is

$$R_e = \frac{Lv\rho}{\mu} = 271849.32$$

Substituting values of R_e & M in equation [12.2],

$$C_f = 0.0058$$

Form factor for vertical tail is calculated from equation [12.3].From section 8.7, We know that the airfoil used in the wing is NACA0012. For this airfoil, x/c = 0.3, t/c = 0.12. Sweep angle at maximum thickness line is λ, it is calculated by equation [12.4]. Here, $\lambda_{1/4} = 0^o$, AR = 2.96, TR = Taper ratio = 1, n = 0. Hence, $\lambda = 0^o$. Substituting values of λ, M, x/c and t/c in equation [12.3]

$$FF_{VT} = 1.0129$$

From section 3.2, $\frac{S_{VT}}{S_{ref}}$ = 2.003(0.1215/1.08) = 0.2253
For Vertical tail, Q = 1.03
Substituting these values in equation [12.1]

$$C_{D_{VT}} = 0.0014$$

12.1.4 Fuselage

For fuselage, the characteristic dimension is its Length (L) = 1.6 m, velocity (v) = 20 m/s, density of air (ρ) = 1.225 kg/m^3 and viscosity coefficient of air (μ) = 1.825 $\times 10^{-5}$ Pa.s. Hence, corresponding Reynold's numbers is

$$R_e = \frac{Lv\rho}{\mu} = 2147945.21$$

Substituting values of R_e & M in equation [12.2],

$$C_f = 0.0039$$

Form factor for fuselage is calculated as

$$FF_f = 1 + \frac{60}{(l_f/d_f)^3} + \frac{(l_f/d_f)}{400} \tag{12.5}$$

Here, f = $\frac{L}{D_{max}}$ = $\frac{1.6}{0.23}$ = 6.9565
where L and D_{max} are the length and maximum diameter of the fuselage respectively.
Substituting value of f in equation [12.5]

$$FF_f = 1.1956$$

For Fuselage, Q = 1
From the CAD model of the fuselage, $\frac{S_w}{S_{ref}}$ = 1.0944
Substituting these values in equation [12.1]

$$C_{D_f} = 0.0051$$

12.1.5 Landing Gear

The landing gear used in our UAV is similar to Round strut or wire. According to [31] Chapter 12 Table 12.5, D/q(Frontal area of landing gear) = 0.3. From sections 9.4.2 & 9.4.3, Frontal area of landing gear = $\pi(0.155^2$ - $0.145^2)$ + 0.015(0.01) + 0.21(0.015) = 0.01272 m^2.
From section 6.10, Wing reference area = 1.08 m^2

$$C_{D_l} = \frac{D}{q(S_{ref})} = 0.00354$$

12.1.6 Vertical Propellers

According to Propeller Drag [**?**], The drag coefficient of a vertical propeller is

$$C_{D_P} = C_{D_{pp}} + C_{D_m} \tag{12.6}$$

In general, $C_{D_{p0}}$ = 0.01
For vertical propeller,

$$C_{D_{pp}} = C_{D_{p0}} \frac{(ND_{prop}C_{0.7R})}{S_{ref}}$$

Blade diameter (D_{prop}) = 22 inches = 0.5588 m, Number of rotors (N) = 8, S_{ref} = 1.08 m^2, Chord at 0.7R ($C_{0.7R}$) = 0.0737 m (Using image processing software on fig [3.6])

$$\Rightarrow C_{D_{pp}} = 0.00305$$

For motor, the drag coefficient is

$$C_{D_m} = C_{D_{0m}} \frac{NlD}{S_{ref}} \tag{12.7}$$

Here, In general $C_{D_{0m}}$ = 1.2, Length of motor(L) = 68.8 $\times$ 10^{-3} m, Diameter of motor(D) = 39.9 $\times$ 10^{-3} m, Number of motors(N) = 4, S_{ref} = 1.08 m^2

$$\Rightarrow C_{D_M} = 0.0122$$

Substituting the drag coefficients of motor and propeller,

$$C_{D_P} = 0.01525$$

From the above subsections and equation [12.1], Total Parasitic Drag is

$$C_{D_0} = C_{D_w} + C_{D_{HT}} + C_{D_{VT}} + C_{D_f} + C_{D_p} + C_{D_l} = 0.03854$$

12.2 Estimating the drag polar

The drag polar is estimated in this section with the finalised design of UAV is given below from Raymer [31] Chapter 12 (eq 12.47)

$$C_D = C_{Do} + KC_L^2 \tag{12.8}$$

Where K is the constant that is given below,

$$K = \frac{1}{\pi(AR)e} \tag{12.9}$$

Here e is the Oswalds efficiency factor and AR is the aspect ratio of the wing.

The Oswalds efficiency factor for a wing is approximated with the formula from the Raymer [31] Chapter 12 (Eq 12.48) since the sweep angle is low, we approximate the wings with the straight wing formula,

$$e = 1.78\left[1 - 0.045(AR)^{0.68}\right] - 0.64 \tag{12.10}$$

From section 6.3, We know that the aspect ratio of the wing is 12. Substituting this value in equation [12.3]

e = 0.706

Substituting the value of e & AR in equation [12.2]

K = 0.0376

At cruise condition, Lift (L) = Weight = 23.4 $\times$ 9.81 = 229.554 N, density of air (ρ) = 1.225 kg/m^3, velocity (v) = 20 m/s, S_{ref} = 1.08 m^2. The lift coefficient is

$$C_L = \frac{2L}{\rho v^2 S_{ref}} = 0.86755$$

Substituting the values of C_{D_0}, C_L and K in equation [12.7],

$$C_D = 0.06684$$

The drag polar is attached below,

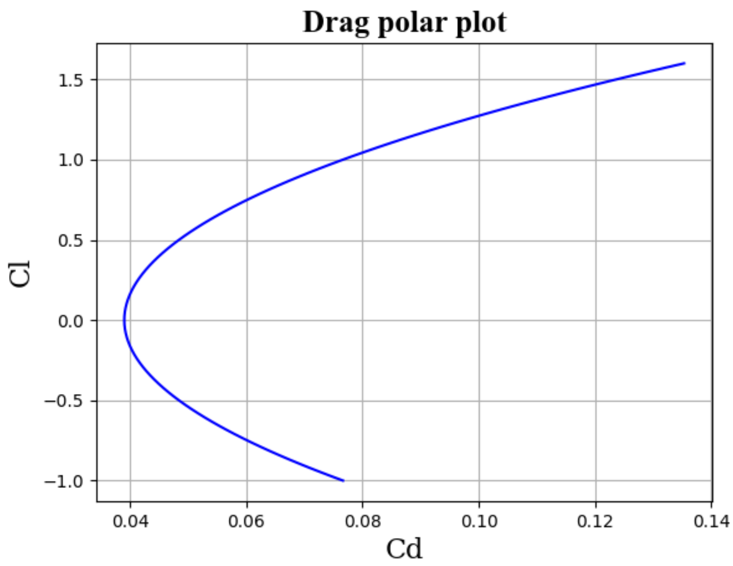


Figure 12.1: Drag polar

12.3 V -N Diagram

The V-n diagram summarises the aircraft's structural and aerodynamic limitations at a particular weight, altitude, and configuration.

$$n = \frac{L}{W} = \frac{\rho S V^2 C_{Lmax}}{2W} \tag{12.11}$$

Considering

$$n_{max} = 1.6 \tag{12.12}$$

Corner velocity is the minimum velocity at which the plane can fly at maximum load factor and not stall.Thus our positive corner velocity for UAV is:

$$v^* = \sqrt{\frac{2n(W/S)}{\rho S C_{Lmax}}} \tag{12.13}$$

$$v^* = 18.83 m/s \tag{12.14}$$

For the cornors velocity without payload, we will reduce 4kgs from the MTOW and calculate the cornor velocity with the same approach. The value of cornor velocity without payload is V^* **= 17.15 m/s**. Stall speed also changes to V_{Stall} **= 13.58 m/s**.
The dive velocity represents the maximum dynamic pressure, exceeding dive velocity may lead to cases like wing divergence and control reversal. It is calculated as follows:

$$V_{dive} = (1.5) V_{cruise} \tag{12.15}$$

$$V_{dive} = 30 m/s \tag{12.16}$$

Considering our cruise velocity as v = 20m/s and C_L = -0.5, when maximum negative angle of attack. The n_{min} can be calculated as follows:

$$n_{min} = \frac{\frac{1}{2}\rho v^2 S C_l}{W} \tag{12.17}$$

$$n_{min} = -0.5 \tag{12.18}$$

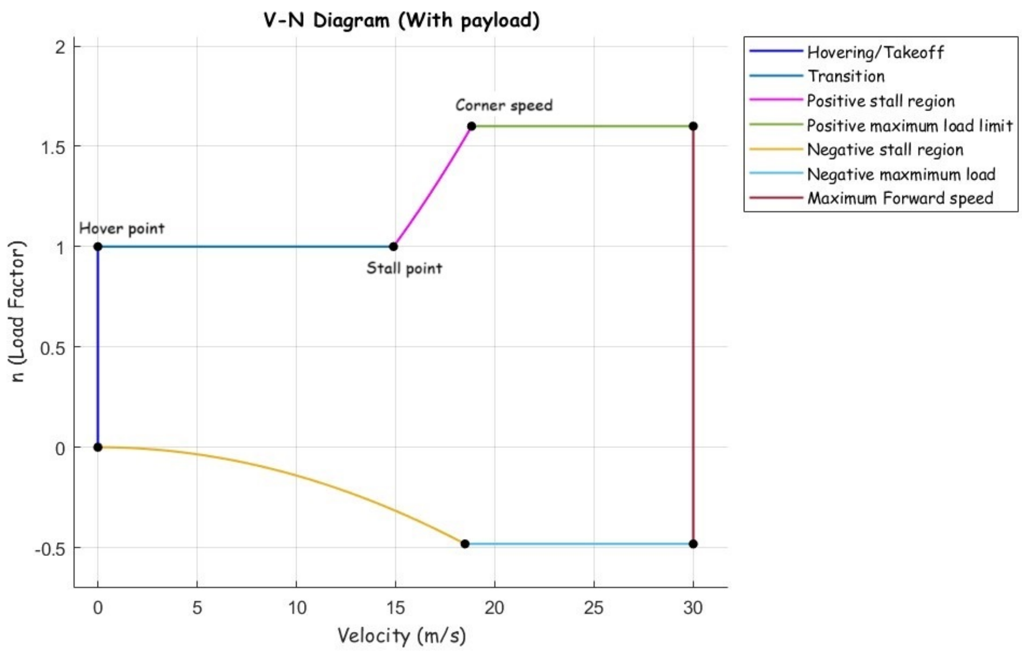


Figure 12.2: V-N diagram with payload

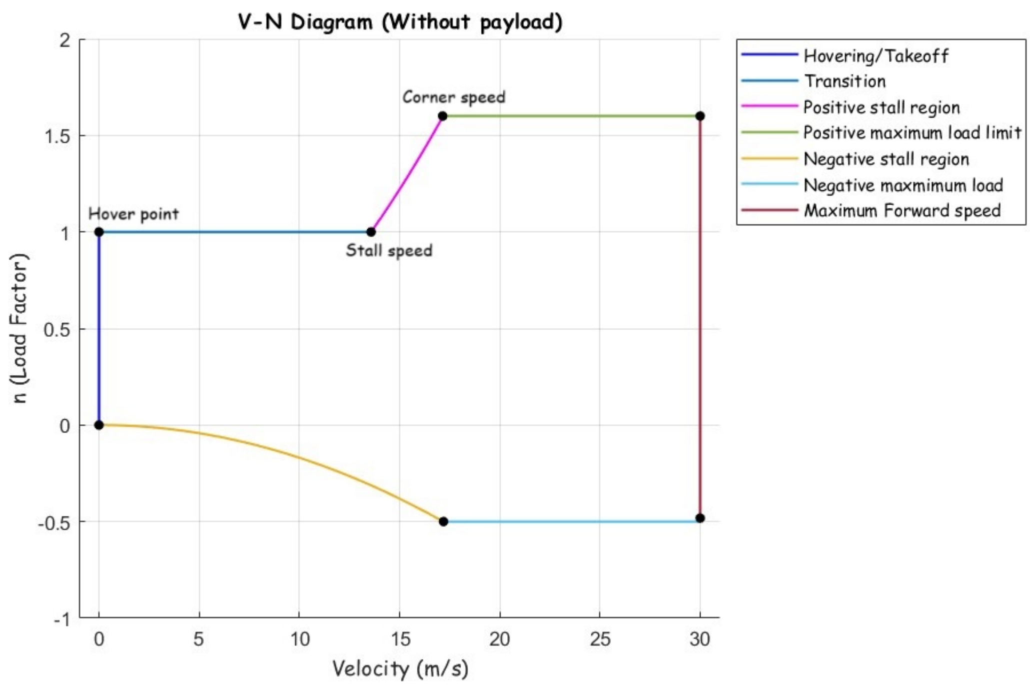


Figure 12.3: V-N diagram without payload

12.4 Range & Endurance

The product of the number of runs and time per run gives the endurance for our UAV. Similarly, range is given by a product of the number of runs and distance per run. The number of runs can be found using the equation,

$$n = \frac{Battery\ energy}{Energy\ per\ run}$$

The battery energy is given by the chosen battery, i.e. 1980 Wh.

Energy per run is calculated by considering the power consumption in each stage of the run. A python code, given in C, is used to calculate it.

$$n = \frac{1980}{586.36} = 3.376$$

Further, the Range is calculated by multiplying the total distance per run, i.e., 50 km x 3.376 = 168.83. Endurance is calculated to be $\frac{(47)(3.376)}{60}$ = 2.65 hrs.

12.5 Three view diagram

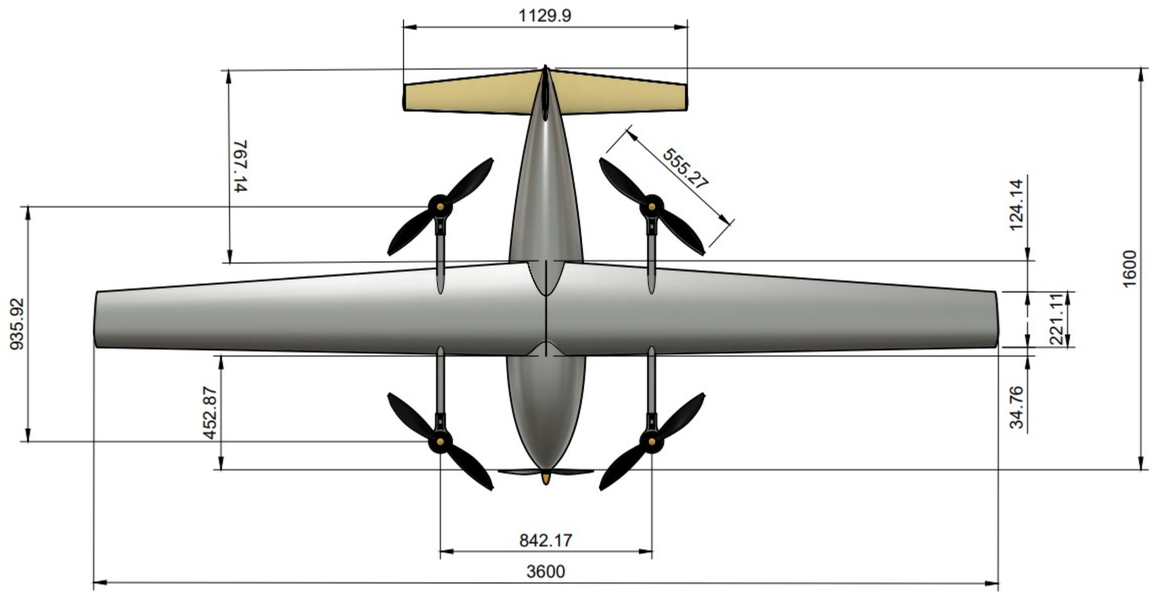


Figure 12.4: Top view

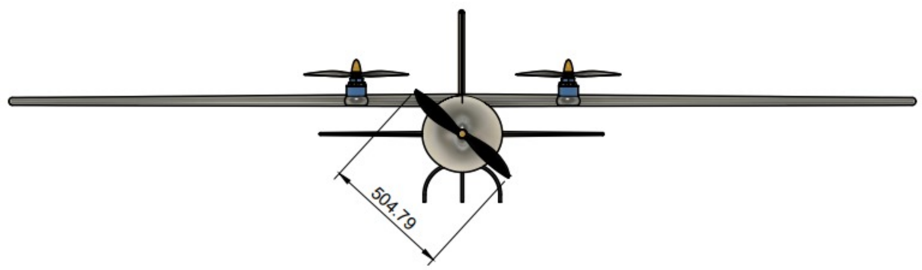


Figure 12.5: Front view

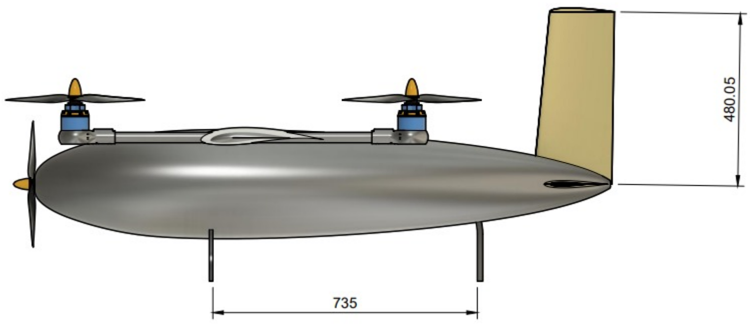


Figure 12.6: Side view

Appendix A

First weight estimation code

A.1 Empty weight fraction curve fitting code

Listing A.1: Empty weight fraction fitting

```
# Importing the libraries needed
import math
import matplotlib.pyplot as plt
import numpy as np

#Dataset import into the code
MTOW = np.array( [18, 30, 30, 15, 13, 23, 35, 15,10 ])
Emp_w = np.array([9.88, 9.09, 12.8, 5.04, 6.5, 9.8, 14.69, 7.15, 4.4])
#Empty weight fraction
we_w0 = Emp_w/MTOW

#Finding the co-efficients from log-log plot
log_MTOW = np.log(MTOW)
log_we_w0 = np.log(we_w0)
#The Val is used to store the values of co-efficient
Val = np.polyfit(log_MTOW,log_we_w0,1)
print(Val)
plt.figure(figsize=(6,5))
MTOWplot=np.linspace(5,35,100)
We_wofit = np.exp(Val[1])*MTOWplot**(Val[0])

#plotting the curve fitted and data
plt.plot(MTOW,we_w0,'.r',markersize=7)
plt.plot(MTOWplot,We_wofit,'-g',linewidth=2.5)
bigfont={'family':'Times New Roman','weight':'heavy','size':'18'}
font = {'family' : 'serif','weight' : 'light', 'size'   : 12}
plt.grid()
plt.xlabel('MTOW',font)
plt.legend(["Data","Fitted"])
plt.title('MTOW vs We/W0',bigfont)
plt.ylabel('We/Wo',font)

#plotting log-log plot
plt.figure(figsize=(5,5))
plt.plot(log_MTOW,log_we_w0,'.g')
bigfont={'family':'Times New Roman','weight':'heavy','size':'18'}
font = {'family' : 'serif','weight' : 'light', 'size'   : 12}
x =np.array([2.2,3.6])
plt.xlabel('log(MTOW)',font)
plt.grid()
plt.ylabel('log(We/Wo)',font)
y = Val[0]*x+Val[1]
plt.plot(x,y,'-b')
plt.legend(["Data","Fitted"])
```

A.2 Code to find estimated weight

This section involves the code for repeated iterations until we converge to find the total weight.

Listing A.2: Importing libraries

```
# Importing the libraries needed
import math
import matplotlib.pyplot as plt
import numpy as np
```

The next part involves the function which gets the empty weight fraction from the total weight passed into the function from the information we got from the previous section.

Listing A.3: Function Empty weight fraction

```
#Returning the value of We/wo From Wo which we fitted...
def We_wo_iter(W0,logA,l):
    return np.exp(logA)*W0**l
```

The next part of the code gives us the total weight from the equation we used to estimate total weight from battery weight, payload weight etc.

Listing A.4: Function Estimating total weight

```
#Total weight which is the Rhs side of the equation
def Totalweight(We_W0,Wold,Wbattery):
    payload =4
    Wb_W0 = Wbattery / Wold
    return payload / (1 - (Wb_W0) - (We_W0))
```

Next part of the code is the important one. Finding the battery weight for our mission requirements with the total weight and payload weight as input.

Listing A.5: Function Estimating total battery weight

```
def Batteryweight(Wold,Wpay):

################### Takeoff energy #####################

    t_takeoff = 70/3600
    v_takeoff = 1.5
    W_takeoff = Wold*9.81
    n_rotors = 4
    K = 1.5   #Thrust to weight ratio
    rho = 1.2 #density of air
    A_prop = 0.1640 #area covered by a propellor (18inch dia)
    TH_takeoff = (K*W_takeoff)/n_rotors

    Power_takeoff_1 = ((TH_takeoff*v_takeoff)/2)*(np.sqrt(1+((2*TH_takeoff)
        /(rho*(v_takeoff**2)*A_prop))))
    Power_takeoff = Power_takeoff_1*n_rotors
    Energy_takeoff = Power_takeoff*t_takeoff

################### Cruise energy ####################

    Cd = 0.07#drag coefficient
    S = 1.0 #span area of wing
    rho = 1.2 #density of air
    V_cruise = 20
    t_cruise = 1250/3600

    Power_cruise_direct = 0.5*rho*(V_cruise**3)*S*Cd
    Power_cruise = Power_cruise_direct / 0.85  #efficiency factor
    Energy_cruise = Power_cruise * t_cruise


```

```
################### Hovering energy #######################
    Fom = 0.7 #Figure of merit
    T_hover_all = W_takeoff*1.1
    T_hover_1 = T_hover_all/4
    t_hover = 70/3600

    Power_hover_1 = (T_hover_1**1.5)/((Fom)*np.sqrt(2*rho*A_prop))
    Power_hover_all = Power_hover_1*4

    Energy_hover = Power_hover_all*t_hover

################### Transition energy #####################
    #from cruise and takeoff
    t_trans =30/3600
    Power_trans = (Power_cruise/8 + Power_hover_all/2)
    Energy_trans = Power_trans*t_trans

################### Landing energy ########################
    Vh = np.sqrt((T_hover_1)/(2*rho*A_prop))
    V_descent = 2.5
    x = - V_descent/Vh
    Vi = (K - 1.125*x - 1.372*(x**2) - 1.718*(x**3) - 0.655*(x**4))*Vh
    t_descent = 40/3600

    Power_landing = K*(Wold-Wpay)*(Vi-V_descent)*4
    Energy_landing = Power_landing*t_descent

################### TOTAL ENERGY ######################

    total_energy_1run = Energy_takeoff*1.5 + Energy_trans*4 + 2*
       Energy_cruise + Energy_landing*1.5 +Energy_hover
    print('Totalenergy', total_energy_1run)

    N_runs = 3.5
    Total_energy = N_runs*total_energy_1run

################### Battery weight #######################
    Battery_energy_density = 240 #WattHr
    Battery_weight = Total_energy/Battery_energy_density
    print("Battery weight: ",Battery_weight)

    return Battery_weight
```

Now the iteration/converging part of the code. Both the methods below depict similar calling process.

Method: Running an iteration to check the error between the next value of total weight to the previous value of total weight and when this converges the iteration is stopped. So in this part we take a initial guess and pass it into the total weight function we get a total weight now finding the error and passing the new value as the intial guess and so on for convergence.

Listing A.6: Iteration1

```
W0old =17
Wpay =4
error = 1
iter=1
iteration=[]
W_weight =[]

while error > 0.0001:
    we_wtotal = We_wo_iter(W0old,Val[1],Val[0])
    Wbat = Batteryweight(W0old,Wpay)
    Wtotal = Totalweight(we_wtotal,W0old,Wbat)
    error = (W0old - Wtotal)**2
```

```
    W_weight.append(W0old)
    iteration.append(iter)
    W0old = Wtotal
    iter+=1
print('The estimated weight: ',W0old,'kg')

plt.figure()
plt.plot(iteration,W_weight,'.-g')
bigfont={'family':'Times New Roman','weight':'heavy','size':'18'}
font = {'family' : 'serif','weight' : 'light', 'size'   : 12}
plt.grid()
plt.xlabel('Iterations',font)
plt.title('Weight vs Iterations',bigfont)
plt.ylabel('Weight',font)
```

Here is a flowchart on how does the code work given in this appendix.

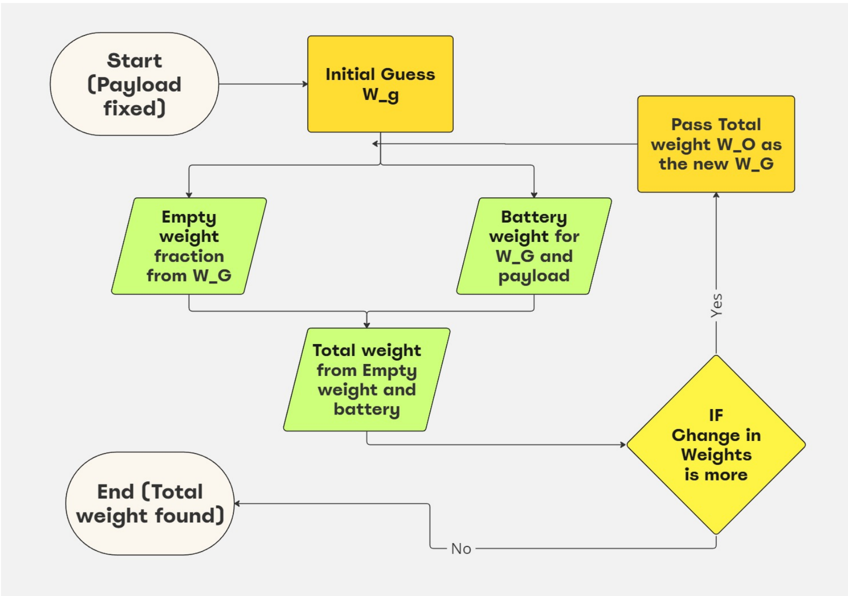


Figure A.1: A flowchart representation of Algorithm

Appendix B

Second weight estimate

For the second estimate we will follow the similar procedure with some distinct changes in the formulation used. Since from Chapter 3 and Chapter 4 we got to estimate some of the parameters of our VTOL. So from that we will make a better weight estimate. The formulations used is given in Chapter 5.

B.1 A walk through of the code

Initially we will be adding all the libraries needed. The code is written in python.

Listing B.1: Importing necessary libraries

```
import math
import matplotlib.pyplot as plt
import numpy as np
```

Next step we will import the value of curve fitted for the empty weight fraction as a function of total weight from chapter 2.

Listing B.2: Weight fraction function

```
Fitted_val = [-0.15325411, 0.6684033034153476] #From the curve fitted

#return empty weight
def We_w0_sec(W0):
    return Fitted_val[1]*(W0**Fitted_val[0])
```

Now we will define the total weight function to return the total weight function from other weight fraction. Also from the weight fraction of powerplant we have calculated in Chapter 5, eqn(**??**), we will find the airframe weight fraction by subtracting the powerplant weight fraction from the empty weight fraction. Also returns the airframe weight.

Listing B.3: Returning airframe weight fraction

```
#Finding the fraction of propellor and motor weight of our VTOL
Wpm_W0 = 3.8/23.5 #Powerplant weight fraction estimated from previous
    reports.

#Total weight which is the Rhs side of the equation
def Totalweight_new(We_W0,Wbattery):
    payload =4
    #Finding the airframe weight fraction from the powerplant weight
        fraction and empty weight fraction
    Wa_W0 = We_W0-Wpm_W0
    if (Wa_W0<1):
        return (payload + Wbattery + 3.8) / (1 - (Wa_W0)),Wa_W0
```

Now we will define the battery weight estimation according to our mission segments with revised parameters given in table(**??**).

Listing B.4: Battery weight estimation with revised parameters

```
def Batteryweight_new(Wold,Wpay):
    Battery_weight = 8.8 #as obtained from previous reports.
    return Battery_weight
```

Now with the necessary functions defined we will run the iteration to decide/converge on the weight.

Listing B.5: Iteration to converge on the weight

```
W0old =20 #initial weight estimate
Wpay =4 #payload weight
error = 1
iter=1
iteration=[]
W_weight =[]
WA_W0=[]
batw =[]
while error > 0.0001:
    we_wtotal = We_w0_sec(W0old)
    Wbat = Batteryweight_new(W0old,Wpay)
    Wtotal,Wa_W0 = Totalweight_new(we_wtotal,Wbat)
    error = (W0old - Wtotal)**2
    batw.append(Wbat)
    WA_W0.append(Wa_W0)
    W_weight.append(W0old)
    iteration.append(iter)
    W0old = Wtotal
    iter+=1
    print('The estimated weight: ',W0old)

#plotting the MTOW vs iterations
plt.figure()
plt.plot(iteration,W_weight,'o-g')
bigfont={'family':'Times New Roman','weight':'heavy','size':'18'}
font = {'family' : 'serif','weight' : 'light', 'size'  : 12}
plt.grid()
plt.xlabel('Iterations',font)
plt.title('Weight vs Iterations',bigfont)
plt.ylabel('Weight',font)

#Plotting the Airframe weight fraction vs iterations
plt.figure()
plt.plot(iteration,WA_W0,'o-b')
bigfont={'family':'Times New Roman','weight':'heavy','size':'18'}
font = {'family' : 'serif','weight' : 'light', 'size'  : 12}
plt.grid()
plt.xlabel('Iterations',font)
plt.title('Airframe weight fraction vs Iterations',bigfont)
plt.ylabel('Airframe weight fraction',font)

#Plotting the battery weight vs iterations
plt.figure()
plt.plot(iteration,batw,'o-r')
bigfont={'family':'Times New Roman','weight':'heavy','size':'18'}
font = {'family' : 'serif','weight' : 'light', 'size'  : 12}
plt.grid()
plt.xlabel('Iterations',font)
plt.title('Battery weight vs Iterations',bigfont)
plt.ylabel('Battery weight',font)
```

Appendix C

Range and Endurance code

This section of the code is to find the range and endurance of the VTOL with all the parameters set in.

C.1 A walk through

Initially we will be adding all the libraries needed. The code is written in python.

Listing C.1: Importing necessary libraries

```
import math
import matplotlib.pyplot as plt
import numpy as np
```

Now we will define a function which calculates the total energy required for a run, and since we have the battery specs defined we will get to know how much runs and how long will the VTOL will be in operation. The mission segments are already defined the chapters.

Listing C.2: Range and endurance for a flight weight

```
# Finding the range and endurance

# Finding the energy per run
def Range_endurance(W_total):

    # Fixed parameters from our report( since our battery is 45000 mAh and
        at 44 volts...
    Batt_energy = (45000*44)/1000
    Wpay = 4
################### Takeoff energy #####################
    t_takeoff = 70/3600
    v_takeoff = 1.5
    W_takeoff = W_total*9.81
    n_rotors = 4
    K = 1.5    #Thrust to weight ratio
    rho = 1.2 #density of air
    A_prop = 0.2451 #area covered by a propellor (22inch dia)
    TH_takeoff = (K*W_takeoff)/n_rotors

    Power_takeoff_1 = ((TH_takeoff*v_takeoff)/2)*(np.sqrt(1+((2*TH_takeoff)
        /(rho*(v_takeoff**2)*A_prop))))
    Power_takeoff = Power_takeoff_1*n_rotors
    Energy_takeoff = Power_takeoff*t_takeoff
    #print('Power takeoff: ',Power_takeoff)
    print('Energy takeoff: ',Energy_takeoff)

################### Cruise energy ####################
    Cd = 0.07#drag coefficient
```

```
    S = 1.08 #span area of wing
    rho = 1.2 #density of air
    V_cruise = 20
    t_cruise = 1250/3600

    Power_cruise_direct = 0.5*rho*(V_cruise**3)*S*Cd
    Power_cruise = Power_cruise_direct / 0.85  #efficiency factor
    Energy_cruise = Power_cruise * t_cruise

    #print('Power cruise: ',Power_cruise)
    print('Energy cruise',Energy_cruise)

################### Hovering energy #######################
    Fom = 0.7 #Figure of merit
    T_hover_all = W_takeoff*1.1
    T_hover_1 = T_hover_all/4
    t_hover = 70/3600

    Power_hover_1 = (T_hover_1**1.5)/((Fom)*np.sqrt(2*rho*A_prop))
    Power_hover_all = Power_hover_1*4

    Energy_hover = Power_hover_all*t_hover

    #print('Power hover',Power_hover_all)
    print('Energy hover',Energy_hover)

################### Transition energy #####################
    #from cruise and takeoff
    t_trans =30/3600
    Power_trans = (Power_cruise/8 + Power_hover_all/2)
    Energy_trans = Power_trans*t_trans

    #print('Power trans',Power_trans)
    print('Energy trans',Energy_trans)

################### Landing energy ########################
    Vh = np.sqrt((T_hover_1)/(2*rho*A_prop))
    V_descent = 2.5
    x = - V_descent/Vh
    Vi = (K - 1.125*x - 1.372*(x**2) - 1.718*(x**3) - 0.655*(x**4))*Vh
    t_descent = 40/3600

    Power_landing = K*(W_total-Wpay)*(Vi-V_descent)*4
    Energy_landing = Power_landing*t_descent

    #print('Power landing',Power_landing)
    print('Energy landing',Energy_landing)

################### TOTAL ENERGY #######################

    total_energy_1run = Energy_takeoff*1.5 + Energy_trans*4 + 2*
        Energy_cruise + Energy_landing*1.5 +Energy_hover
    print('Totalenergy1', total_energy_1run)

    N_runs = Batt_energy/total_energy_1run

    Range = 50*N_runs # in kms
    Endurance = (47*N_runs)/60  #in hrs

    print("Range_attained : ",Range)
    print("Endurance :",Endurance)

    return Range,Endurance
```

Now this function can be called with the weight of VTOL to get the range and endurance for out given flight mission.

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