# AE461 : Aircraft design I

# FINAL REPORT


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REPORT

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# APPENDIX

| Chapters | Content |
| --- | --- |
| Chapter 1 | Mission requirements |
| Chapter 2 | Weight and wing-loading estimation |
| Chapter 3 | Wing configuration |
| Chapter 4 | Tail configuration |
| Chapter 5 | Elevator sizing |
| Chapter 6 | Battery weight estimation |

# Chapter 1: Mission Requirements

1. Min. Payload = 40 kgs of payload
2. Max. Range = 1000 km.
3. Max Takeoff and Landing Runway Length = 500 m
4. Service Ceiling = 11 km

5 Cruise Speed = 400 kmph
6 Propulsion: Jet Engines (Turbofans)

The parameters we choose are:-

Cd0 = 0.017 (by historical values it is between 0.014 and 0.02)
Sfc (specific fuel consumption - cruise) = 2.504350800000000e-04,
Sfc (specific fuel consumption - loiter) = 2.226089600000001e-04
Endurance = 40 min
L/D = 9 (high L/D ratio)
Aspect Ratio = 3 (we decided to make a delta wing, we compared the results for various aspect ratio as choose this value)
Cl_max= 2.5 (by historical value for delta wing)
Cd_0LG=0.009 (0.006-0.012 by historical value)
Cd_0HLDto=0.005 (0.003-0.008 by historical value)
Cl_c=0.4 (0.3-0.5 by historical value)
Cl_flapTo=0.5 (0.3-0.8 by historical value)

ρ=1.225 (general density of air at sea level)
Ceiling = 11 km (by mission requirements)
Thus the sigma value is 0.81

Stall velocity= 25m/s
Vmax = 140 m/s (1.2 Vcruise)
ROC(rate of climb) = 7 (by historical value of delta wings, 3-10)
ROC (at ceiling) = 2.5 (for cruise ceiling of delta wing(military) we took ROC=2.5)
Take of distance = 500m

# Chapter 2: Weight and Wing-loading Estimation

For weight estimation, the flight path was:-


By historical data we took the weight ratio as

Table 3: Typical average segment weight fractions

| No. | Mission segment | $W_{i+1}/W_i$ |
|---|---|---|
| 1 | Taxi and take-off | 0.98 |
| 2 | Climb | 0.97 |
| 3 | Descent | 0.99 |
| 4 | Approach and landing | 0.997 |

Thus W0/W1 = 0.98
W2/W1=0.97
W4/W3 = 0.99
W6/W5= 0.997

For W3/W2 the formula for jet aircraft is:

$$\frac{W_3}{W_2} = e^{-\left(\frac{R^*C}{V^*\,L/D}\right)}$$

Thus W3/W2 = 0.79

And for W5/W4

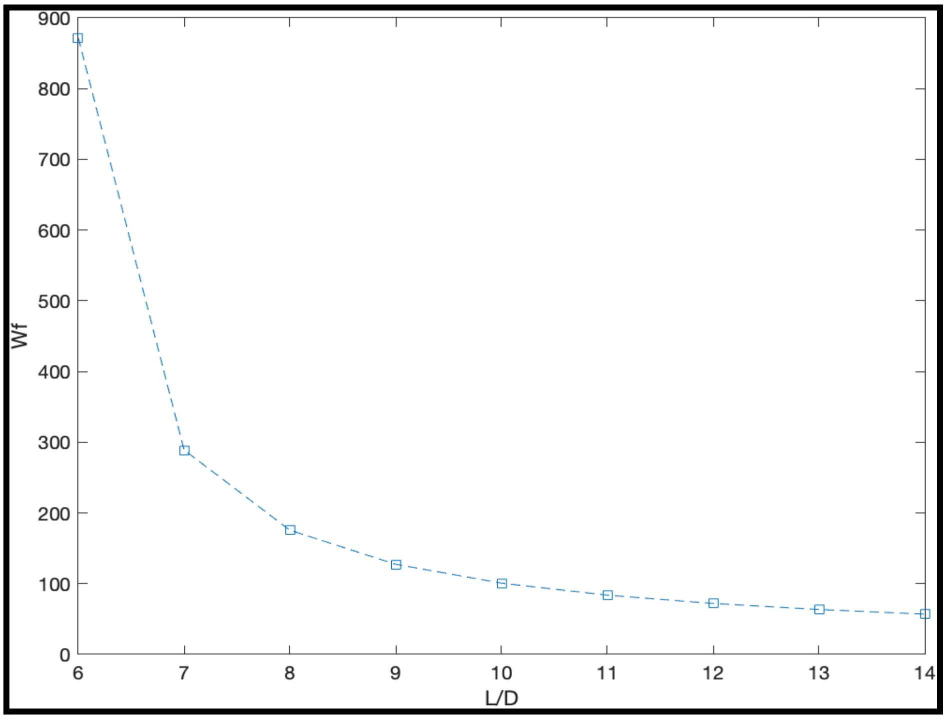
$$\frac{W_5}{W_4} = e^{-\left(\frac{E^*C}{L/D(max)}\right)}$$

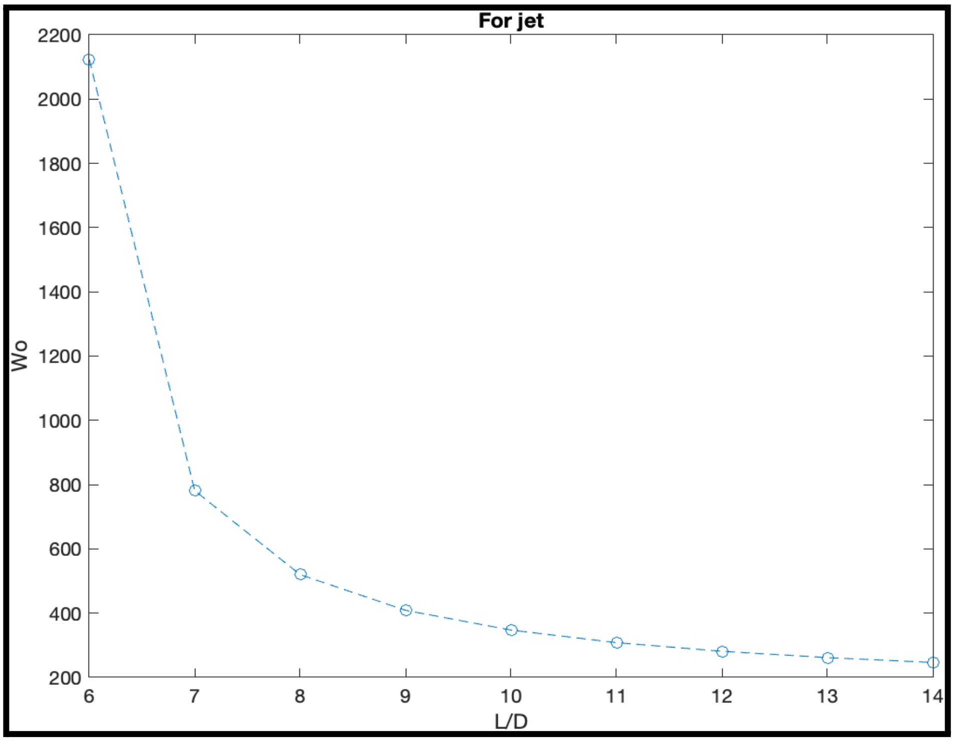
Thus, W5/W4 = 0.94

Thus the W6/W0 = 0.7

$$W_0 = \frac{W_c + W_p}{1 - \frac{W_f}{W_0} - \frac{W_E}{W_0}}$$

Putting in the values we get -






We choose L/D =9 and for which got the following values

W0= 408 kg

For Wp = 50 kg and Wf= 127.41 kg

W/S and T/W was chosen according to the formulas below for our mission profile-

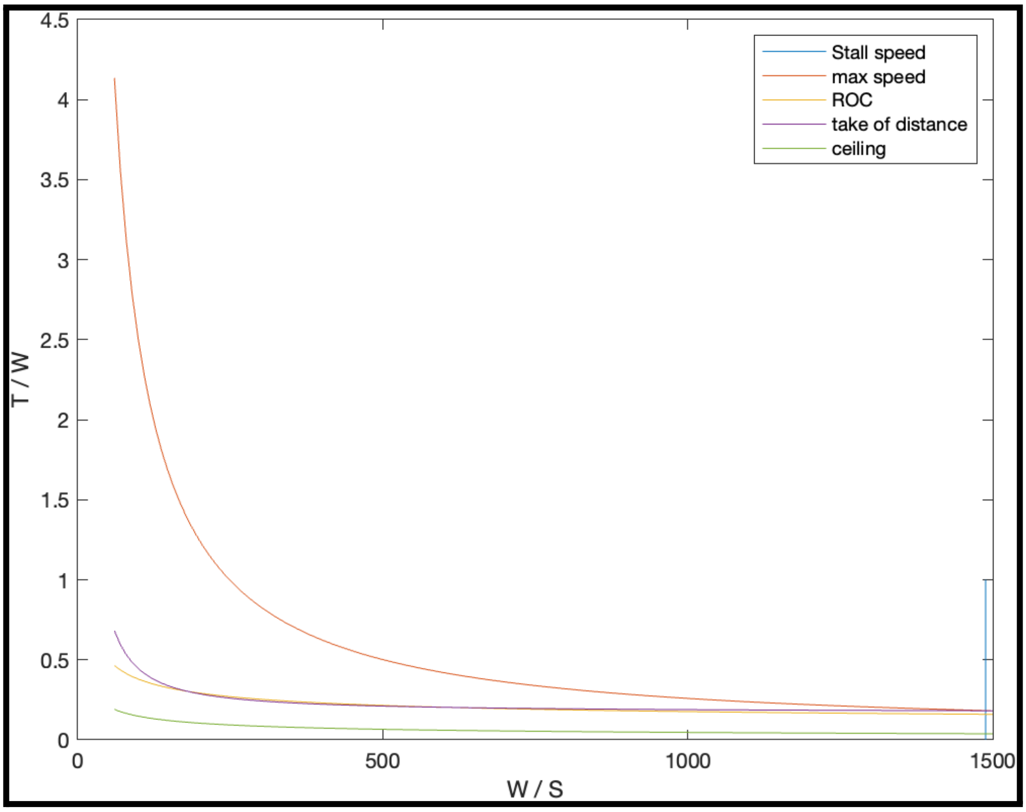
$$\left(\frac{W}{S}\right)_{V_s} = \frac{1}{2}\rho V_s^2 C_{L_{\max}}$$

For maximum velocity:

$$\left(\frac{T_{SL}}{W}\right)_{V_{\max}} = \rho_o V_{\max}^2 C_{D_o} \frac{1}{2\left(\frac{W}{S}\right)} + \frac{2K}{\rho\sigma V_{\max}^2}\left(\frac{W}{S}\right)$$

For rate of climb:

$$\left(\frac{T}{W}\right)_{ROC} = \frac{ROC}{\sqrt{\frac{2}{\rho\sqrt{\frac{C_{D_o}}{K}}}\left(\frac{W}{S}\right)}} + \frac{1}{(L/D)_{\max}}$$




The selection of W/s was very important and we understood and learnt a lot about the designing procedure as these values affected our Wo (as to accommodate our acceptable s and airfoil values we had to make changes again and again n wo as well) and also the are which affected our lab reports 6,7,8 directly and hence to craft a possible design we spent hours getting such value to fall in place and finally the values were chosen and our design point came out to be W/S =1500

# Chapter 3: Wing and airfoil Configuration

Based on our W/S and W0,
our wing planform geometry and airfoil geometry is as follows

- Taper angle:
- Taper ratio=0.1
- Root chord=1.7164 m
- Tip chord= 0.17164 m
- Wing span = 2.8321 m
- Wing area=2.6736 m^2

It will be a delta wing geometry.

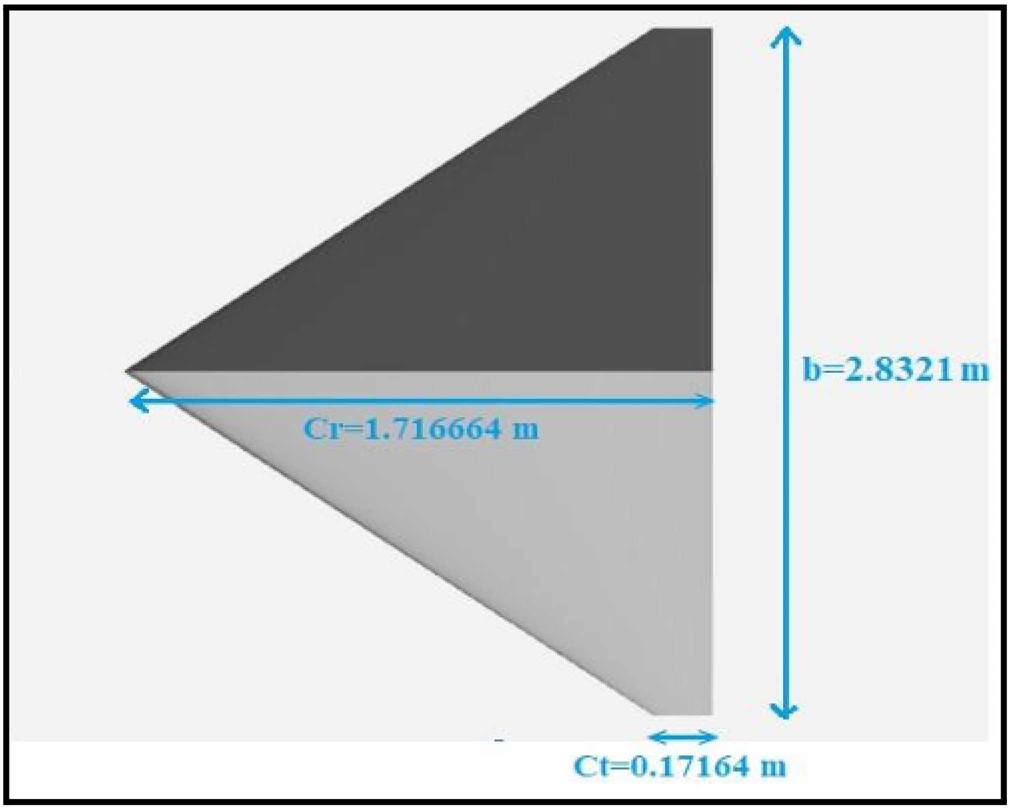
Wing aerofoil estimation

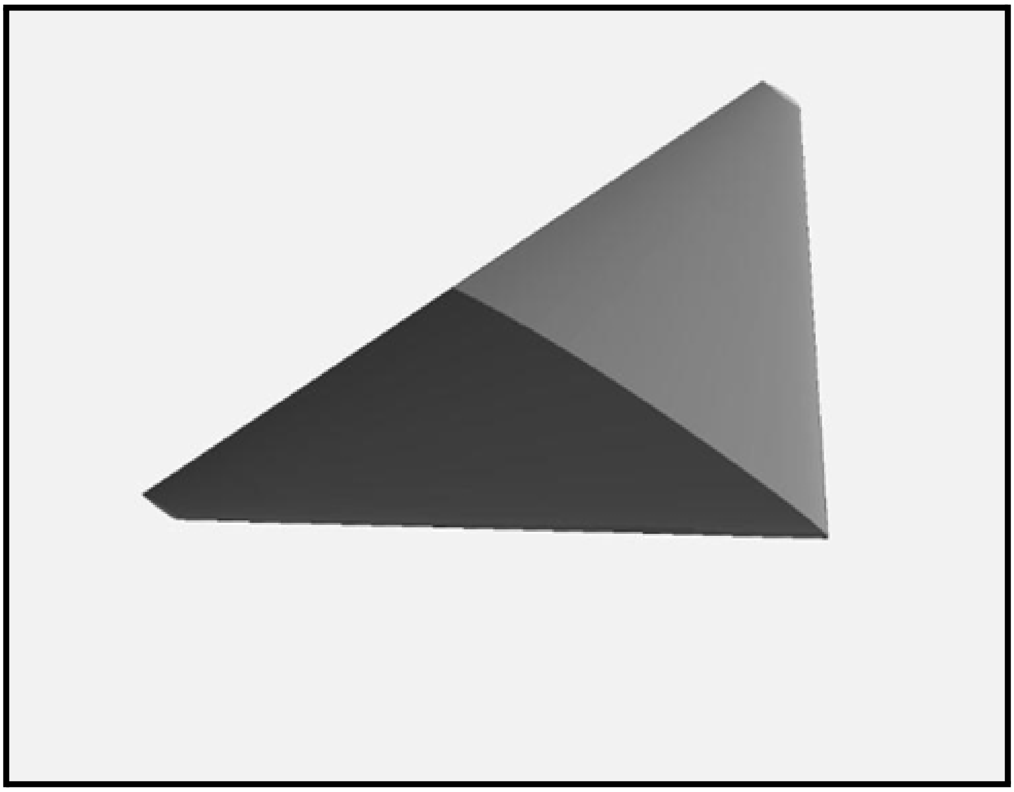
- design lift coefficient.=0.0935
- design angle of attack.=3
- $CL\alpha$,3D=0.101 4. $CL0$=0
- Aerofoil: NACA 16-006 6. L/D= 9 . From the table we calculated for lab 1 given below we choose NACA 0016 for our model.

| 1 | series | reynolds no. | airfoil | Clo | Cl alpha | Cl max | alpho not | alpha stall | Cm ac 2 | Column1 | CLo2 | CL alpha2 | AR |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 2 | 4 | 8 | NACA0006 | 0 | 0.1 | -1 | 0 | 9 | 0 | | 0 | 0.099532542 | 8 |
| 3 | | 3X10^4 | NACA 0009 | 0 | 0.1 | 2 | 0 | 12 | -0.9 | | 0 | 0.099532542 | |
| 4 | | 6 | NACA 0010-34 | 0 | 0.8 | 0.8 | 0 | 11.5 | 0 | | 0 | 0.771030614 | |
| 5 | | 6 | NACA 0012 | 0 | 0.1 | 1.6 | 0 | 16 | 0 | | 0 | 0.099532542 | |
| 6 | | | NACA10-35 | 0 | 0.1 | 0.7 | 0 | 10 | -1.1 | | 0 | 0.099532542 | |
| 7 | | 3X10^4 | NACA 1412 | -1.5 | 0.1 | 1.6 | -1 | 17 | | | 0.099533 | 0.099532542 | |
| 8 | | | NACA1408 | 0.1 | 0.1 | 1.35 | -0.1 | 14 | -0.9 | | 0.009953 | 0.099532542 | |
| 9 | | 3.1 | NACA2408 | 0.2 | 0.1 | 1.5 | -2 | 14 | | | 0.199065 | 0.099532542 | |
| 10 | | | NACA1410 | 0.1 | 0.1 | 1.5 | -0.1 | 14 | -1 | | 0.009953 | 0.099532542 | |
| 11 | | | NACA2410 | 0.2 | 0.09 | 1.6 | -2 | 16 | -1 | | 0.179242 | 0.089621182 | |
| 12 | | 3.1 | NACA2412 | 0.2 | 0.09 | 1.6 | -2 | 16 | -0.2 | | 0.179242 | 0.089621182 | |
| 13 | | 9 | NACA 2415 | 0.2 | 0.1 | 1.6 | -2 | 16 | -0.2 | | 0.199065 | 0.099532542 | |
| 14 | | | NACA4412 | 0.4 | 0.1 | 1.5 | -4 | 14 | -0.4 | | 0.39813 | 0.099532542 | |
| 15 | | | NACA4415 | 0.4 | 0.1 | 1.4 | -4 | 12 | -0.4 | | 0.39813 | 0.099532542 | |
| 16 | | | NACA4418 | 0.4 | 0.1 | 1.4 | -4 | 14 | -0.3 | | 0.39813 | 0.099532542 | |
| 17 | | | NACA4421 | 0.4 | 0.1 | 1.3 | -4 | 12 | -0.3 | | 0.39813 | 0.099532542 | |
| 18 | | | NACA4424 | 0.4 | 0.12 | 1.4 | -4 | 12 | -0.3 | | 0.47731 | 0.119327489 | |
| 19 | 5 | 3*10^4 | NACA 23012 | 0.15 | 0.106 | 1.8 | -1.9 | 18 | -0.5 | | 0.200402 | 0.105474911 | |
| 20 | | 8.9 | NACA 23015 | 1 | 0.1 | 1.63 | -1 | 18 | -0.5 | | 0.099533 | 0.099532542 | |
| 21 | | 6 | NACA 23018 | 0.15 | 0.1 | 1.6 | -1 | 16 | -0.1 | | 0.099533 | 0.099532542 | |
| 22 | | 5.9 | NACA 23021 | 0.1 | 0.1 | 1.5 | -1.3 | 15 | 0 | | 0.129392 | 0.099532542 | |
| 23 | | 8.9 | NACA 23024 | 0.2 | 0.1 | 1.4 | -1 | 15 | 0 | | 0.099533 | 0.099532542 | |
| 24 | 6 | 9 | NACA 63-206 | 0.2 | 0.1 | 1 | -2 | 12 | | | 0.199388 | 0.099693851 | 15 |
| 25 | | | NACA 66-210 | 0.15 | 0.104 | 1.3 | -1.75 | 11 | -0.14 | | 0.181421 | 0.103668909 | |
| 26 | | | NACA 64A140 | 0.35 | 0.105 | 1.6 | -2.75 | 14 | -0.32 | | 0.287822 | 0.104662522 | |
| 27 | | | NACA 64A010 | 0 | 0.102 | 1.225 | 0 | 12 | 0 | | 0 | 0.101681502 | |
| 28 | | | NACA $64_2$-015 | 0 | 0.1 | 1.4 | 0 | 15 | 0 | | 0 | 0.099693851 | |
| 29 | | | NACA $64_2$-215 | 0.17 | 0.1125 | 1.5 | -1.75 | 15 | -0.11 | | 0.196197 | 0.112112678 | |
| 30 | | | NACA $64_2$-415 | 0.35 | 0.166 | 1.65 | -3 | 16 | -0.275 | | 0.495474 | 0.165158076 | |
| 31 | | | NACA 64,-018 | 0 | 0.106 | 1.5 | 0 | 17 | 0 | | 0 | 0.105656074 | |
| 32 | | | NACA 64,-218 | 0.1 | 0.116 | 1.45 | -1 | 17 | -0.125 | | 0.115588 | 0.115588247 | |
| 33 | | 6 | NACA 66-006 | 0 | 0.1 | 0.8 | 0 | 11.8 | 0 | | 0 | 0.099693851 | |
| 34 | | 9 | NACA 66-009 | 0 | 0.1 | 1.05 | 0 | 10.2 | 0 | | 0 | 0.099693851 | |
| 35 | | 6 | NACA 66-206 | 0.2 | 0.1 | 1 | -2 | 11 | -0.15 | | 0.213299 | 0.106649565 | |
| 36 | | 6 | NACA 66-209 | 0.1 | 0.107 | 1.15 | -1.2 | 11 | -0.12 | | 0.119633 | 0.099693851 | |
| 37 | | 9 | NACA $66_1$-012 | 0 | 0.1 | 1.25 | 0 | 14 | 0 | | 0 | 0.099693851 | |
| 38 | | 9 | NACA $66_1$-212 | 0.15 | 0.1 | 1.45 | -1.8 | 15 | -0.12 | | 0.23307 | 0.129483082 | |
| 39 | | 6 | NACA $66_2$-015 | 0 | 0.13 | 1.35 | 0 | 16.2 | 0 | | 0 | 0.159217695 | |
| 40 | | 9 | NACA $66_2$-215 | 0.15 | 0.16 | 1.5 | -1 | 16.3 | -0.12 | | 0.159218 | 0.159217695 | |
| 41 | | 6 | NACA $66_2$-415 | 0.25 | 0.16 | 1.6 | -2.9 | 18 | -0.29 | | 0.289112 | 0.099693851 | |
| 42 | | 5.9 | NACA $66_3$-018 | 0 | 0.1 | 1.3 | 0 | 17 | 0 | | 0 | 0.159217695 | |

As per the data we got Clo value 0 corresponding to an acceptable Cla value and hence a symmetric airfoil was chosen as mentioned above.

The airfoil was chosen with rigorous effort as for our configurations Cl design wasn't coming in an acceptable range but for a symmetric airfoil we got all falling together as the Cl design was in coherence with acceptable alpha design which was 3 and hence we were sure to use a symmetric airfoil



# Chapter 4: Tail/Canard Configuration

As we have delta wing , we do not have tail instead we have canard mounted in front of the main wing and hence the equations were not readily available so we derived it, for Cl and for Cm , as proof and the equations we used are provided below.

down wash = 0

$V_c / V_w = 1$

$$\tfrac{1}{2}\rho V_w^2 S_w \{C_{L_0} + C_{L\alpha}\} = \tfrac{1}{2}\rho V_c^2 S_c \{C_{L\alpha_c}(\alpha + i)\} + \tfrac{1}{2}\rho V_w^2 S_w \{C_{L\alpha_w}\{\alpha\}\}$$

$$C_{L_0} + C_{L\alpha} = \frac{S_c}{S_w}\{C_{L\alpha_c}(\alpha + i)\} + C_{L\alpha_w}\alpha$$

$$\boxed{\begin{aligned} C_{L_0} &= \frac{S_c}{S_w} \times C_{L\alpha_c} i \\ C_{L\alpha} &= \frac{S_c}{S_w} C_{L\alpha_c} + C_{L\alpha_w} \end{aligned}} \quad \text{---(1)}$$

$$\tfrac{1}{2}\rho V_w^2 S_w c_w \{C_{m_0} + C_{m\alpha}\alpha\} = \tfrac{1}{2}\rho V_c^2 S_c \{C_{L_{0c}} + C_{L\alpha_c}(\alpha + i)\} \times (x_{cg} - x_{acc}) - \tfrac{1}{2}\rho V_w^2 S_w \{C_{L\alpha_w}\alpha\}\{x_{acw} - x_{cg}\}$$

$$C_{m_0} + C_{m\alpha}\alpha = \frac{S_c}{S_w} \times C_{L\alpha_c} i\,(\bar{x}_{cg} - \bar{x}_{acc}) + \left\{\frac{S_c}{S_w} C_{L\alpha_c}(\bar{x}_{cg} - \bar{x}_{acc}) - C_{L\alpha_w}(\bar{x}_{acw} - \bar{x}_{cg})\right\}\alpha$$

$$\boxed{\begin{aligned} C_{m_0} &= \frac{S_c}{S_w} \times C_{L\alpha_c} i\,(\bar{x}_{cg} - \bar{x}_{ac}). \\ C_{m\alpha} &= \frac{S_c}{S_w} C_{L\alpha_c}(\bar{x}_{cg} - \bar{x}_{acc}) - C_{L\alpha_w}(\bar{x}_{acw} - \bar{x}_{cg}) \end{aligned}}$$

$$S_t = \frac{\boldsymbol{S_t}}{\boldsymbol{S}} * \boldsymbol{s}$$

$$C_{L_{\alpha_t}(2D)} = \frac{C_{L_{\alpha_t}(3D)}}{1 - \dfrac{C_{L_{\alpha_t}(3D)}}{\pi e_t AR_t}}$$

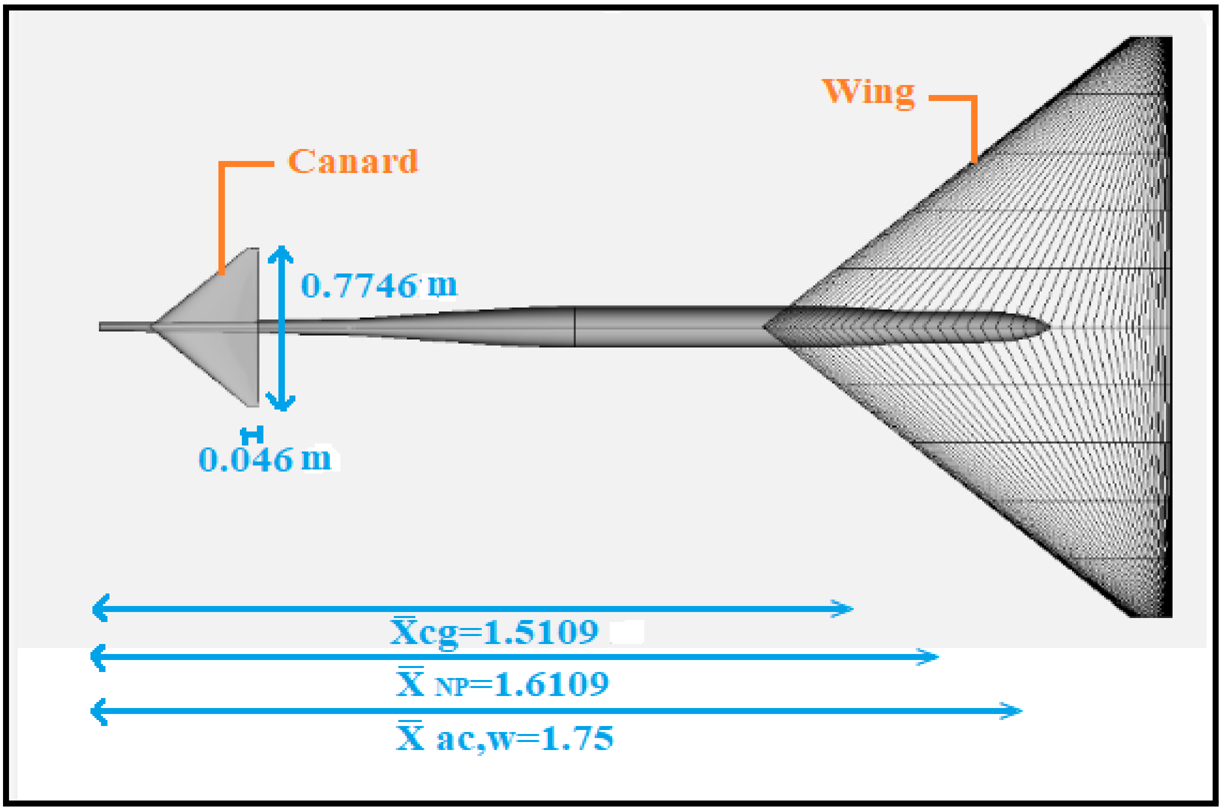
We choose static margin as 10% of the root chord of the main wing and the distance values provided below are normalised with the root chord itself.

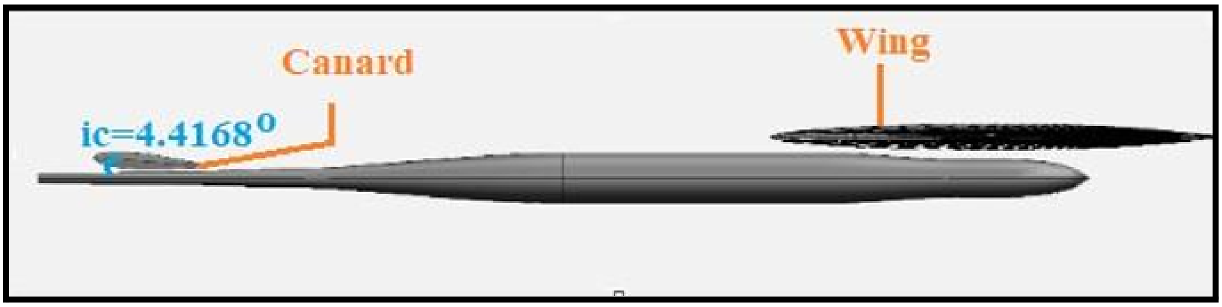
Using the formulas above we got the following:

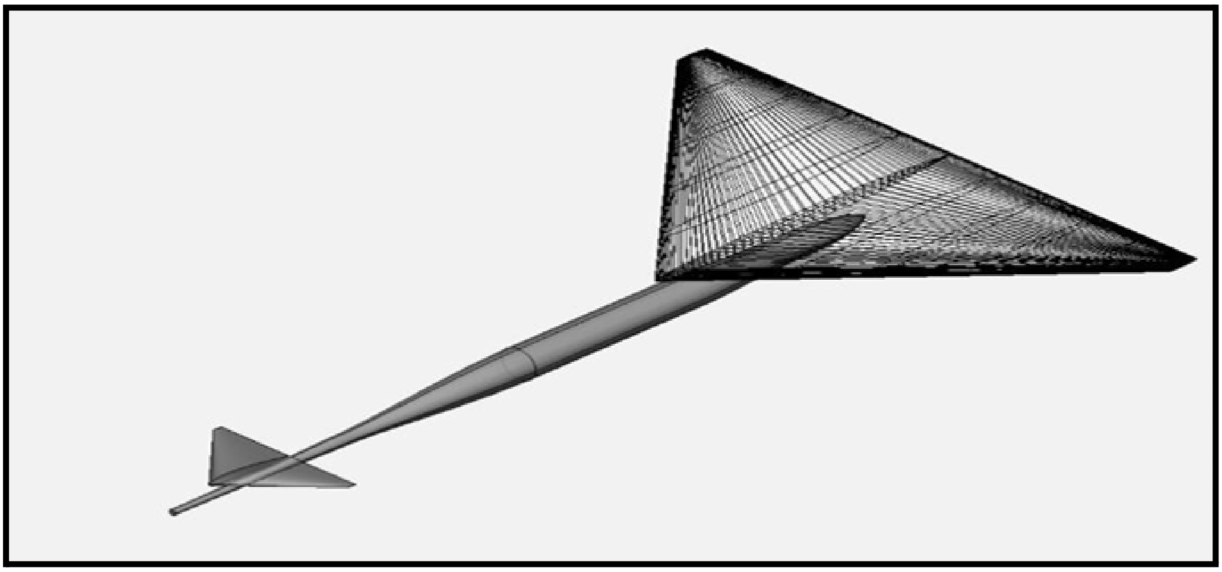
- XNP=1.6109
- Clα=0.10172
- SM=0.1
- Xcg=1.51
- Cm0=0.053822
- ic=4.4168°
- VH=-0.1725 (our formation is canard, equation are above)

- CLα3D=0.11
- Sc=0.2 m^2
- Sc/S= 0.0748
- Selected airfoil= S8035
- Crc=0.46 m
- Ctc=0.046m
- bc=0.7746
- $\lambda$=0.1
- CLα2D=0.11138

And prepared a CAD model describing the same configuration.



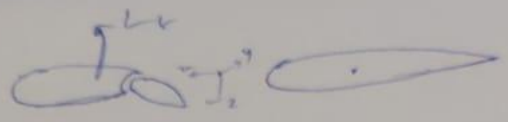


S8035 symmetrical airfoil was chosen for the canard.

# Chapter 5: Elevator Sizing

We got all the parameters like Cm0, Cm_alpha, Static Margin…. As our plane is canard configuration the formula of Cl_deltae and Cm_deltae was little bit changed as following.



$$\Delta L = \Delta L_t$$

$$S \tfrac{1}{2}\rho V_\infty^2 \Delta C_L = \tfrac{1}{2}\rho V_\infty^2 S_t \Delta C_{L_t}$$

$$\Delta C_L = \eta \frac{S_t}{S} \frac{\partial C_{L_t}}{\partial \alpha_t} \frac{\partial \alpha_t}{\partial \delta e} \Delta \delta e$$

$$\therefore \boxed{\frac{\partial C_L}{\partial \delta e} = \eta \frac{S_t}{S} C_{L\alpha_t} \tau}$$

$$\Delta M = \Delta L_t (x_{c.g} - x_{a.c})$$

$$S_t \tfrac{1}{2}\rho V_\infty^2 (\bar{c}) \Delta C_m = \tfrac{1}{2}\rho V_\infty^2 S_t \Delta C_{L_t} (x_{cg} - x_{a.c})$$

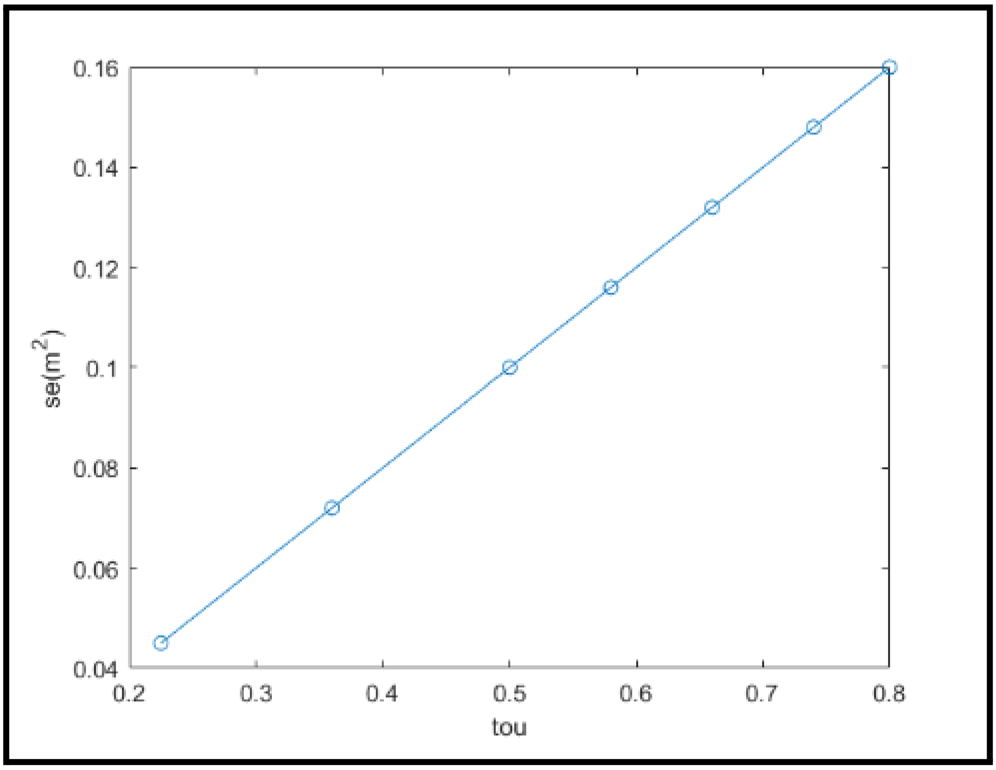
$$\Delta C_m = \eta \frac{S_t}{S} (x_{cg} - x_{ac}) \frac{\Delta C_{L_t}}{\Delta \alpha_t} \frac{\partial \alpha_t}{\partial \delta e} \Delta \delta e$$

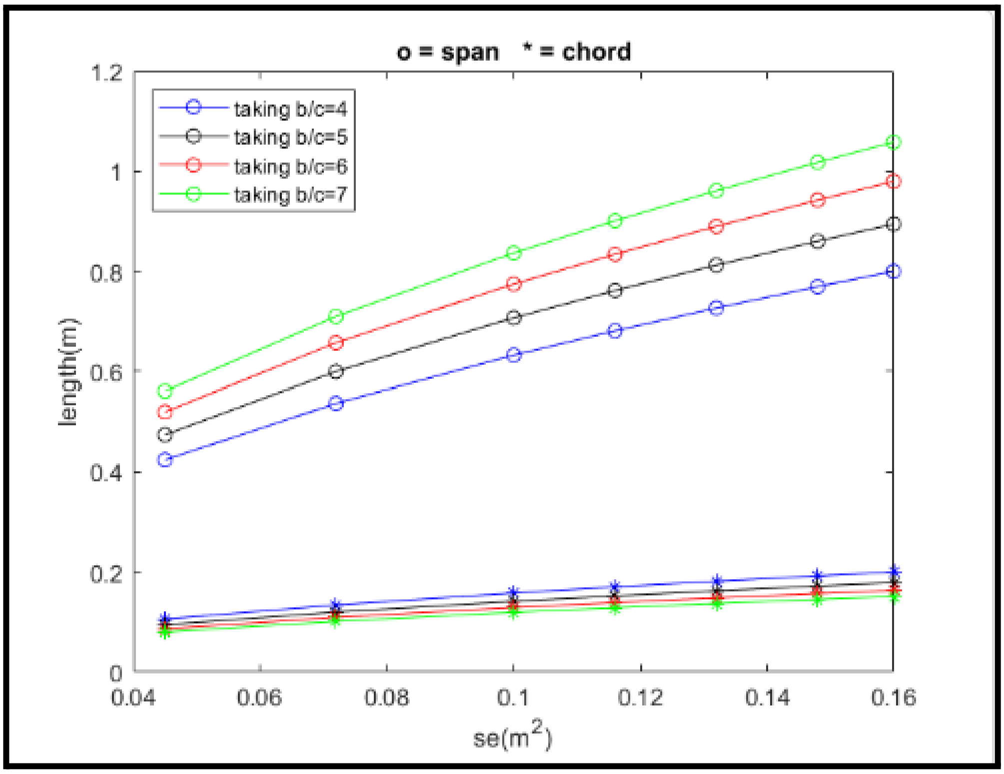
$$\boxed{\frac{\partial C_m}{\partial \delta e} = \eta \left(\frac{S_t}{S}\right) (x_{c.g} - x_{a.c}) C_{L\alpha_t} \tau}$$

Thus for tao= 0.225,0.36,0.5,0.58,0.66,0.74,0.8 and St=0.2m^2

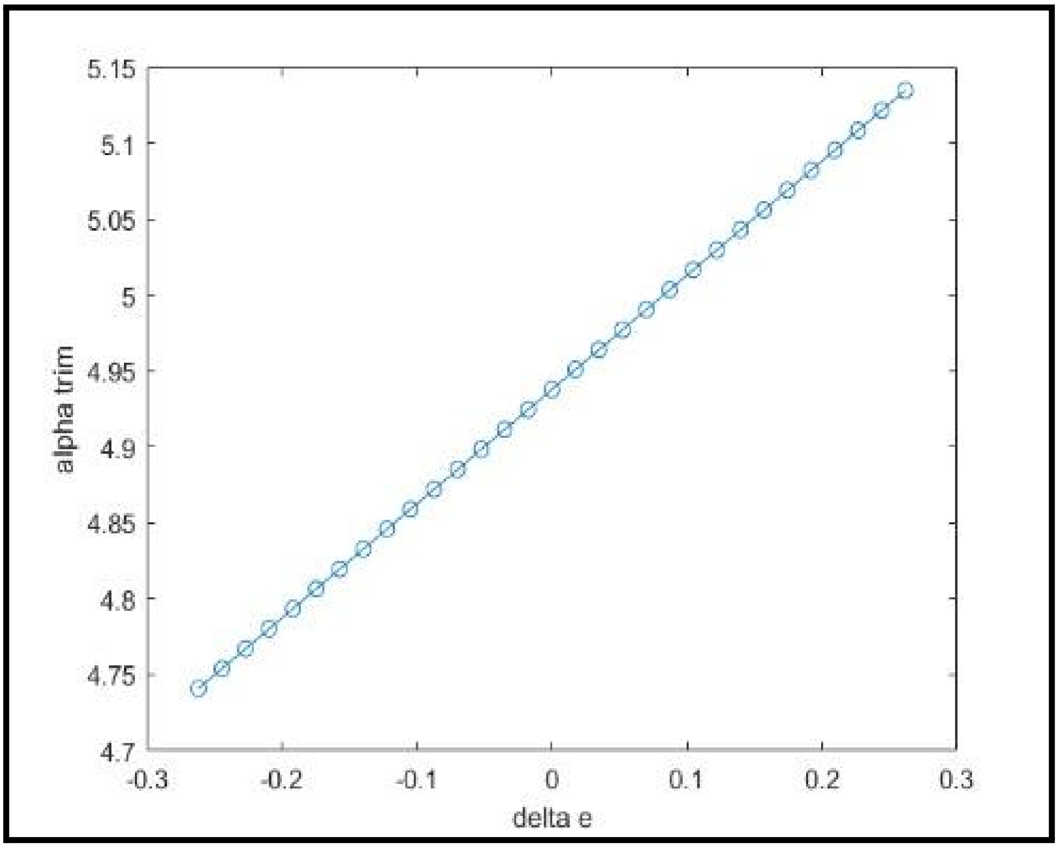
Se came out to be 0.0450, 0.0720, 0.1000, 0.1160, 0.1320, 0.1480, 0.1600

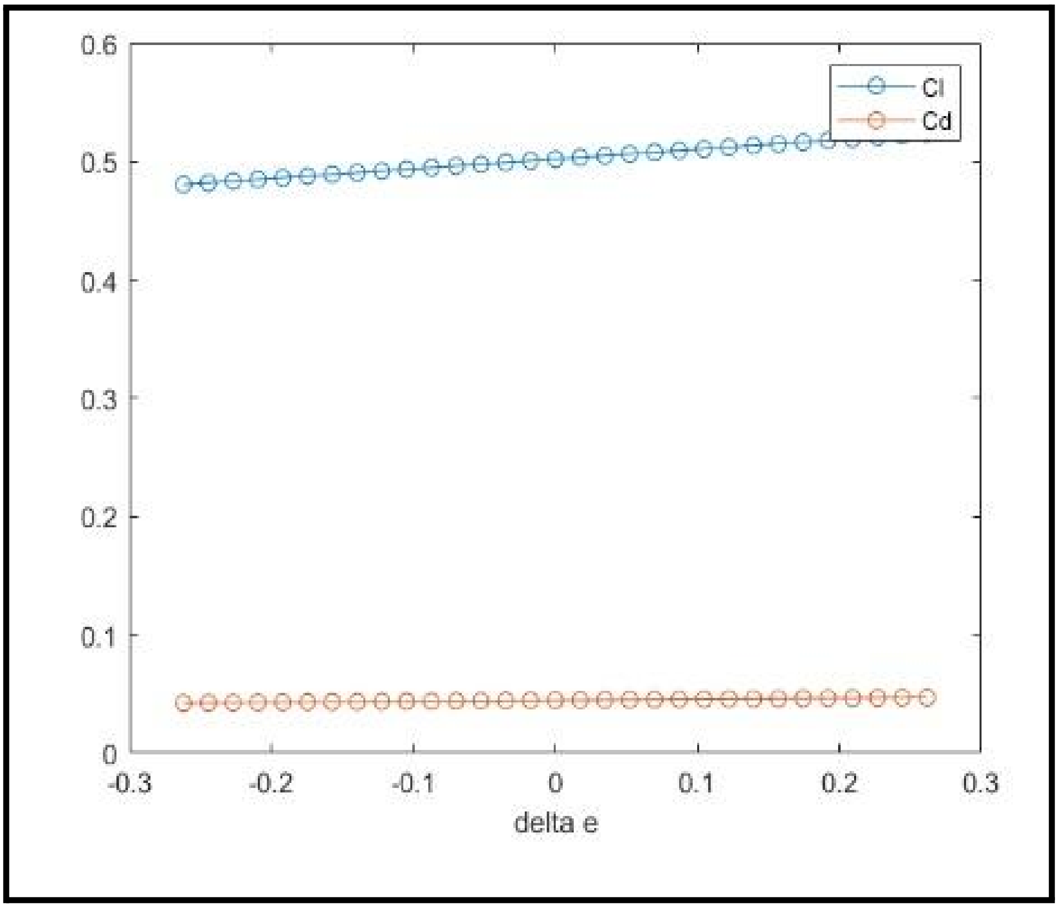
From this we choose Se=0.045 and for which b= 0.42 ,c=0.1

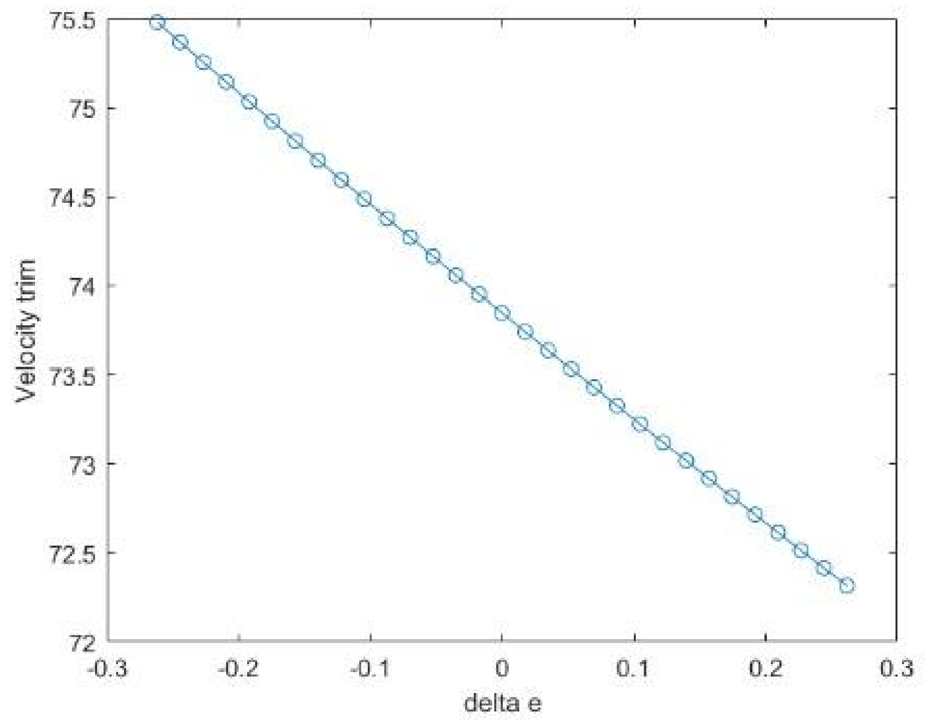


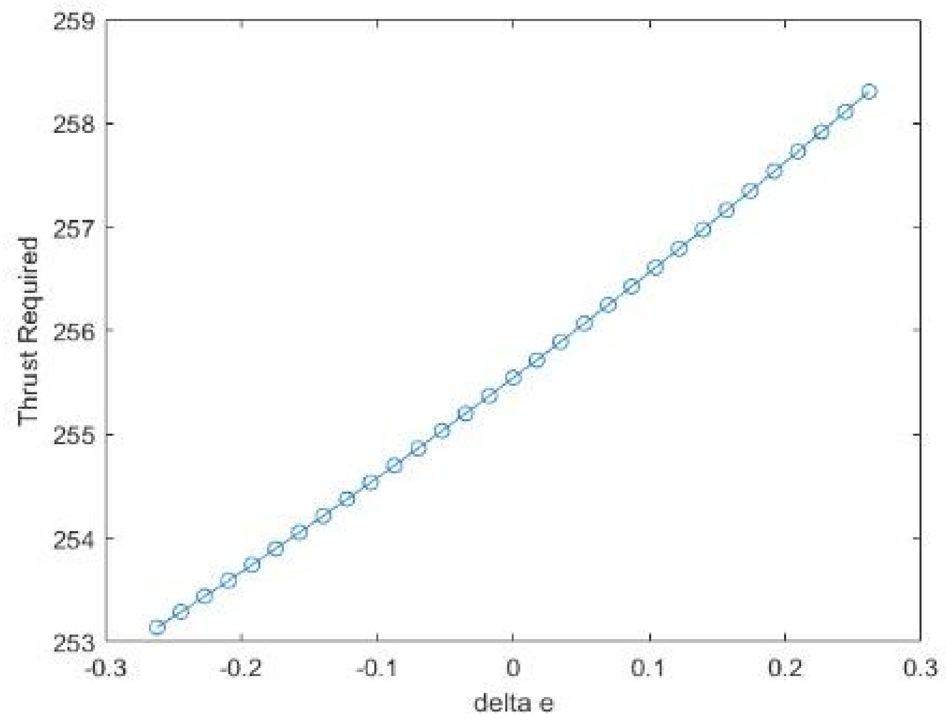

The trim performance for various delta e is:-


5.15
5.1
5.05
5
4.95
4.9
4.85
4.8
4.75
4.7
alpha trim
-0.3
-0.2
-0.1
0
0.1
0.2
0.3
delta e

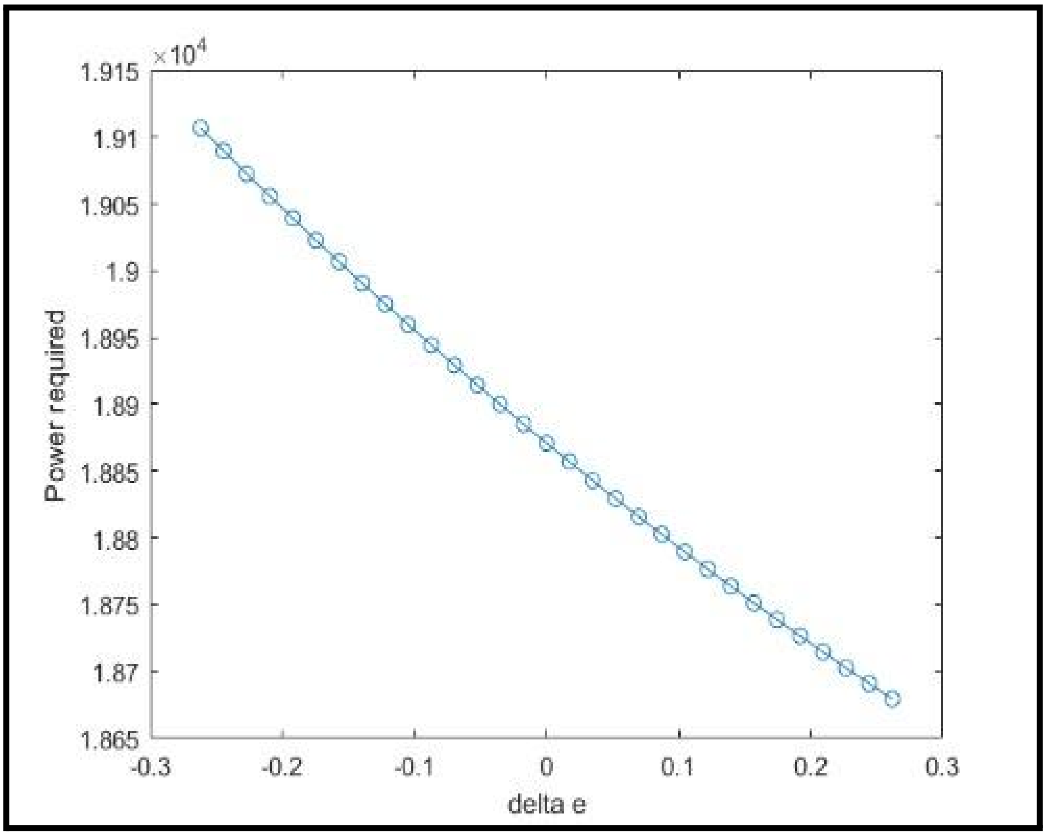

0.6
0.5
0.4
0.3
0.2
0.1
0
Cl
Cd
-0.3
-0.2
-0.1
0
0.1
0.2
0.3
delta e



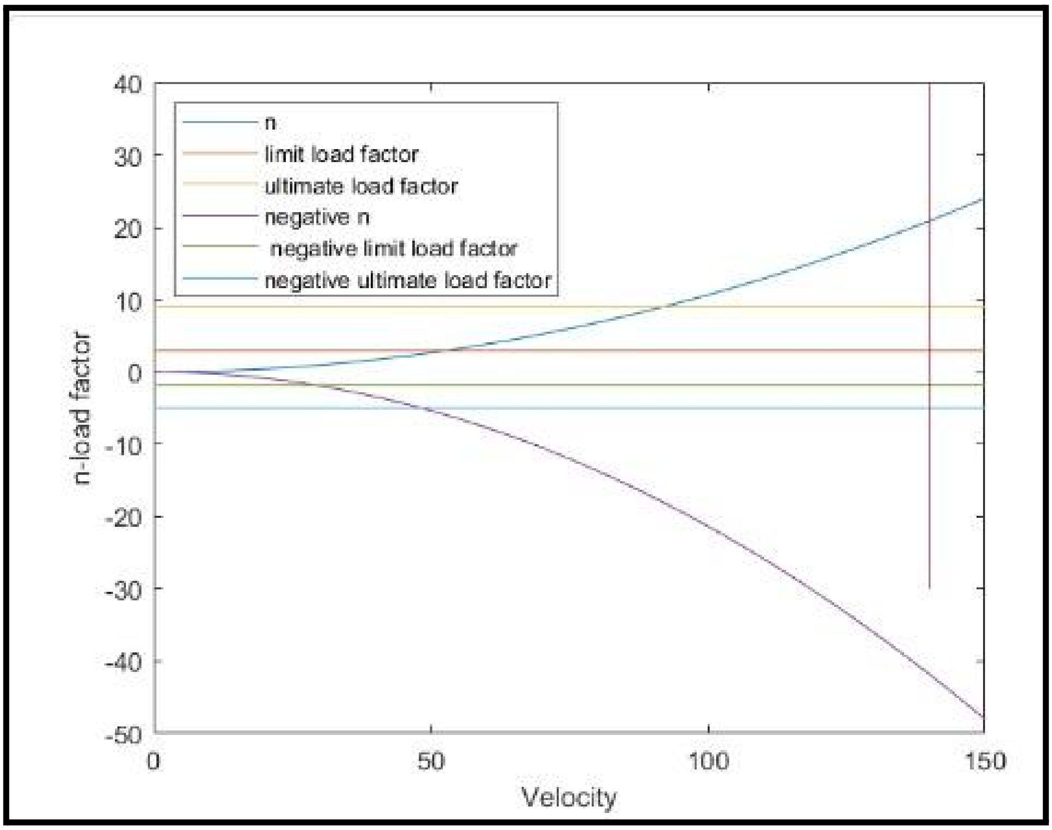
75.5
75
74.5
74
73.5
73
72.5
72
Velocity trim
-0.3
-0.2
-0.1
0
0.1
0.2
0.3
delta e

259
258
257
256
255
254
253
Thrust Required
-0.3
-0.2
-0.1
0
0.1
0.2
0.3
delta e

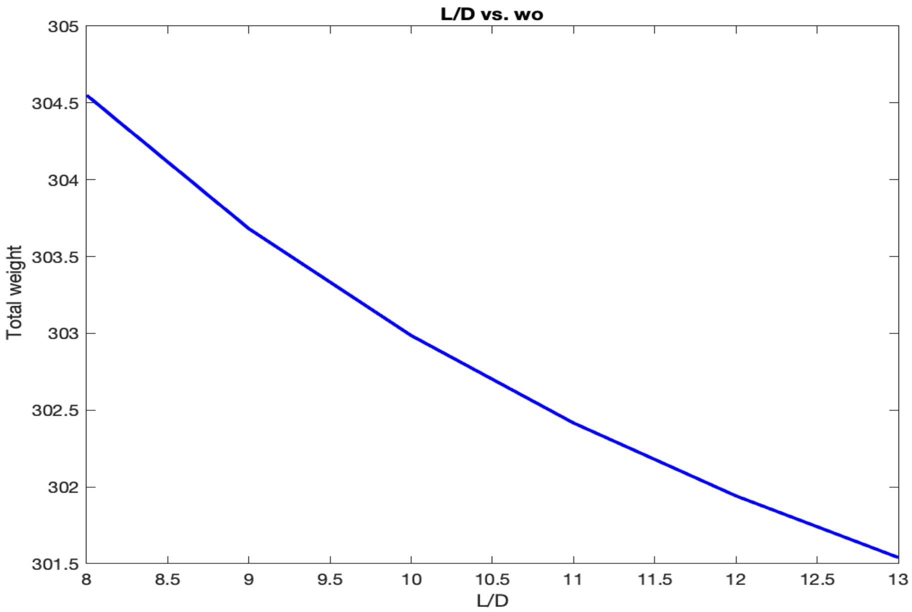


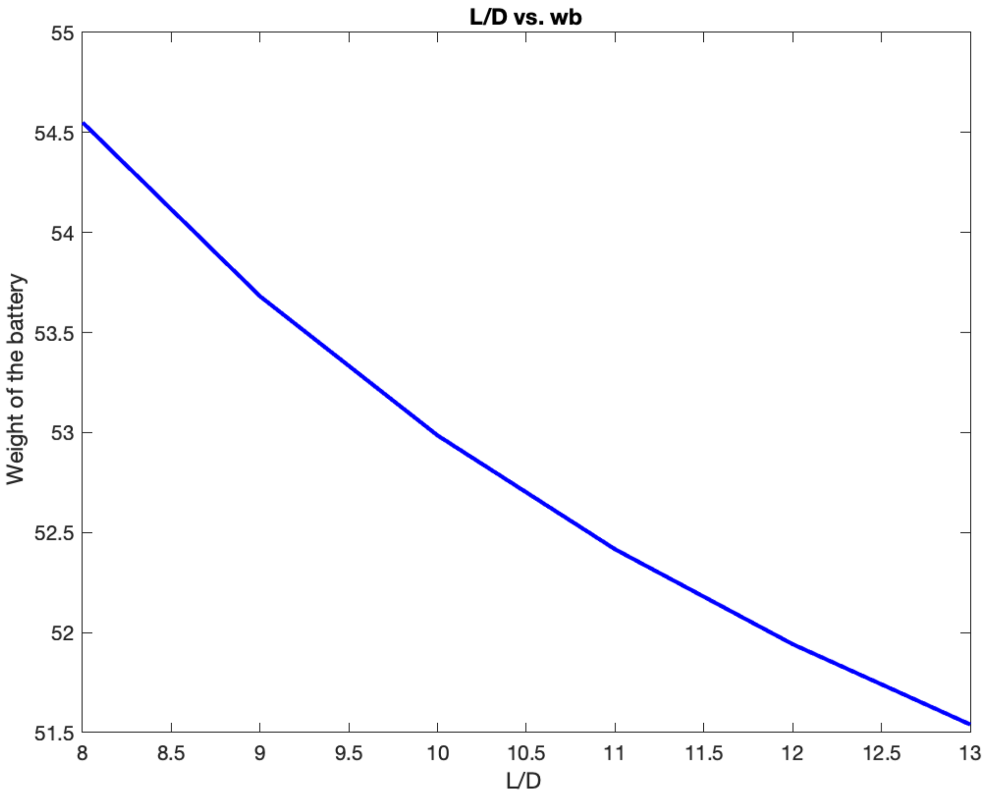

Finally we made the V-n diagram which look like:-



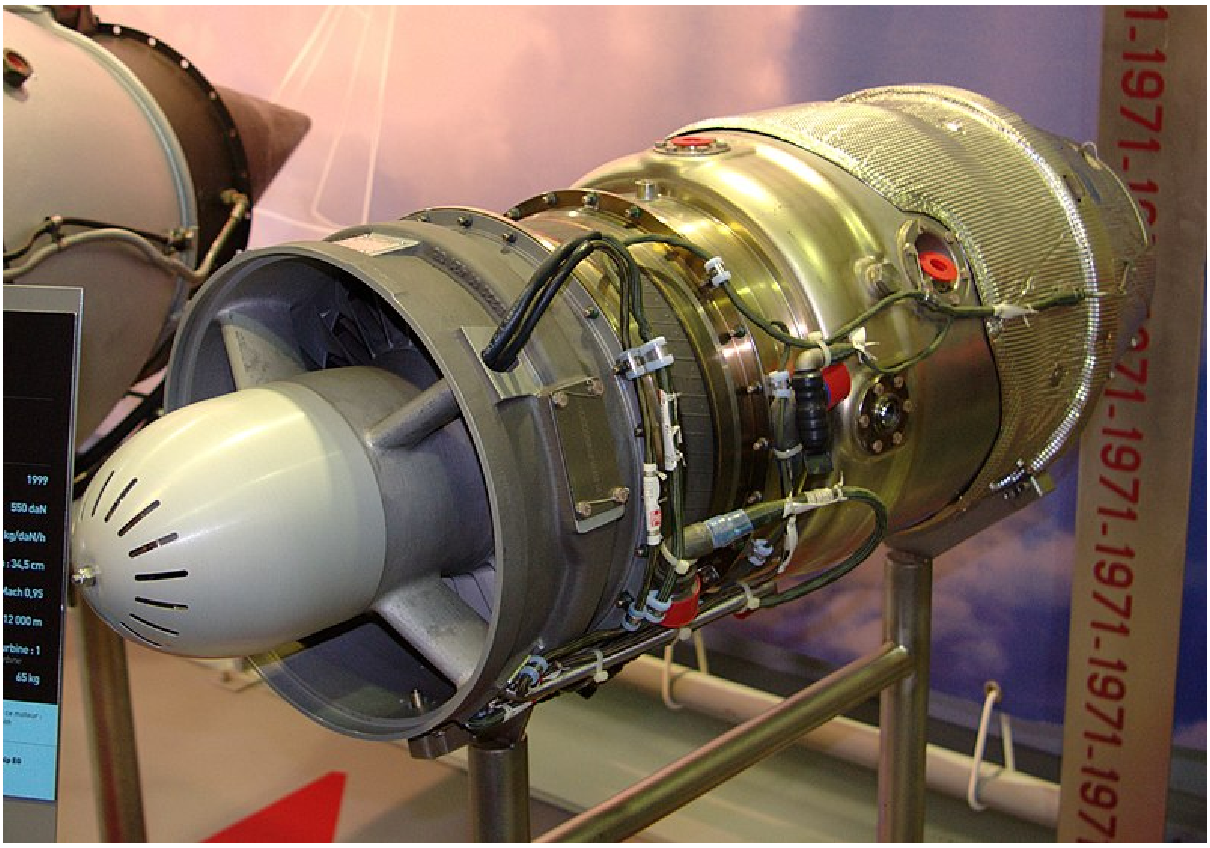
## Chapter 6: Battery weight estimation

For our earlier mission requirements we calculated battery weight using initial guess for battery and we ran the code until we got the change in total weight less than 1e-6. We also considered the changes in the density with the variation in the altitude and hence here are the final values for our mission profiles and



For our model, we took the TRI -60 Microturbo jet engine. It gives a maximum thrust of 3500N which falls under our design parameters




THANK YOU FOR TEACHING US HOW TO DESIGN A PLANE

WE GENUINELY LEARNT THE ACTUAL PROCEDURE ON HOW TO DESIGN A PLANE AND GOT THE FLAVOURS WHAT ARE THE ACTUAL CHALLENGES AND HOW TO RE-ITERATE THE PROCEDURE AND TO NARROW DOWN TO ONE SINGLE VALUE AND DESIGN