AE244
Assignment 3

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22B0003

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Wing Design

Wing Parameters

Parameter	Value
Airfoil	NACA 0010
Taper Ratio	0.25
Angle of Attack	0^o
Twist Angle	0^o
Sweep Angle	4^o
Dihedral Angle	4^o
Span	53.37
Aspect Ratio	21.3

Table 1: Wing Parameters

Plots

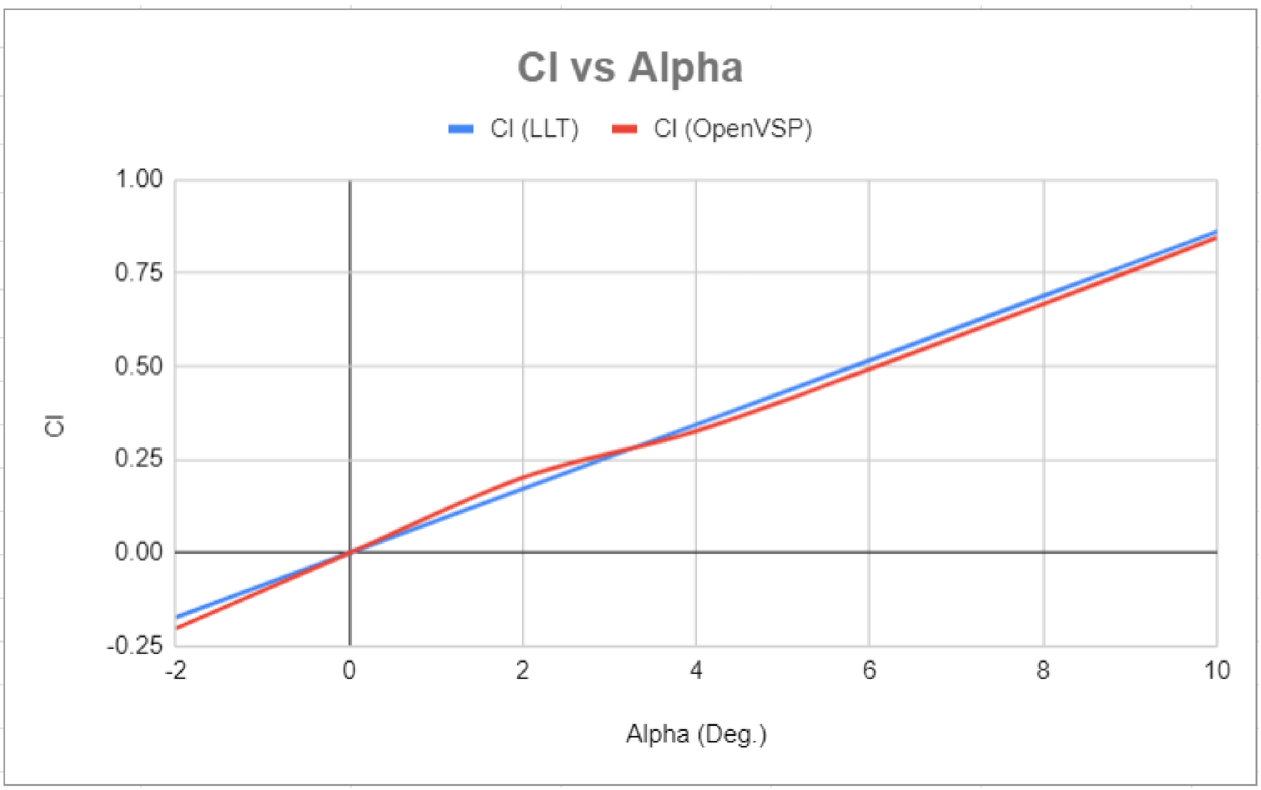


Figure 1: Cl vs Alpha Plots calculated using a) Lifting Line Theory b) OpenVSP

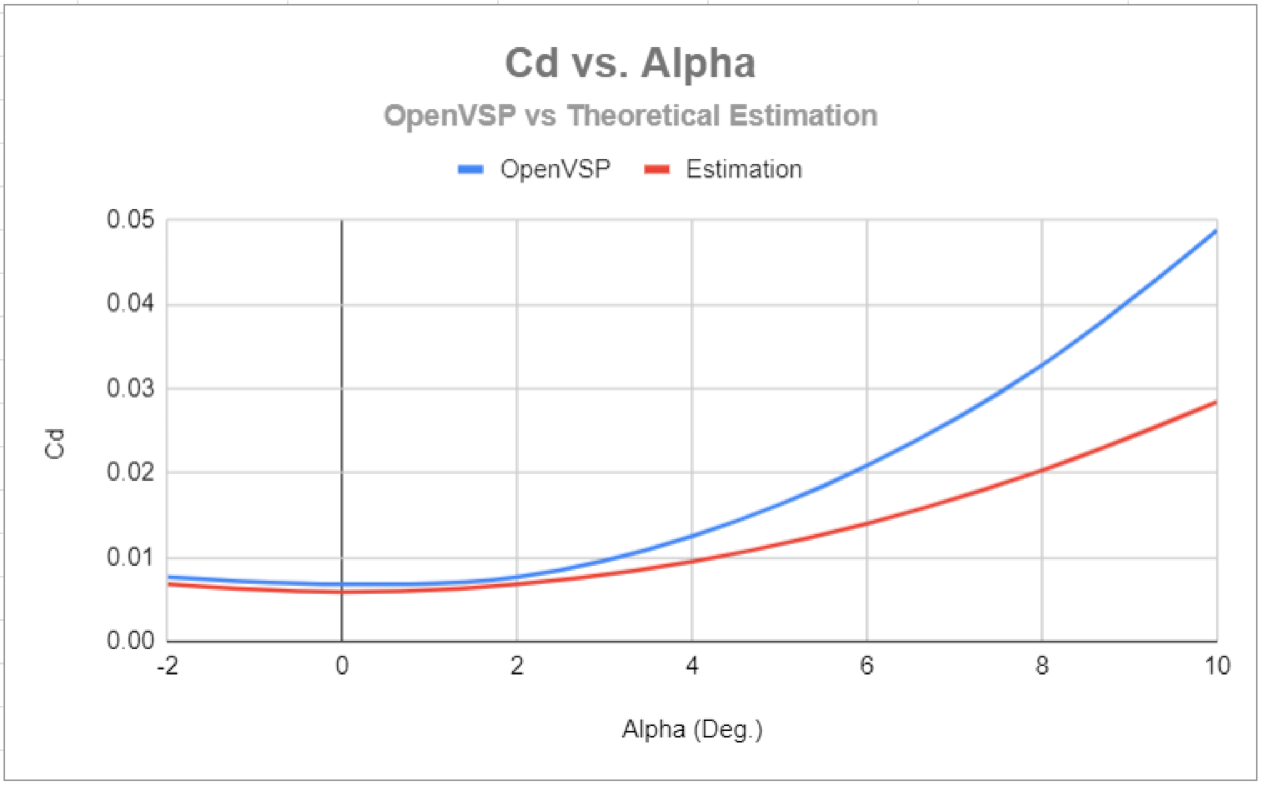


Figure 2: Cd vs Alpha Plots calculated using a) Drag Estimation b) OpenVSP

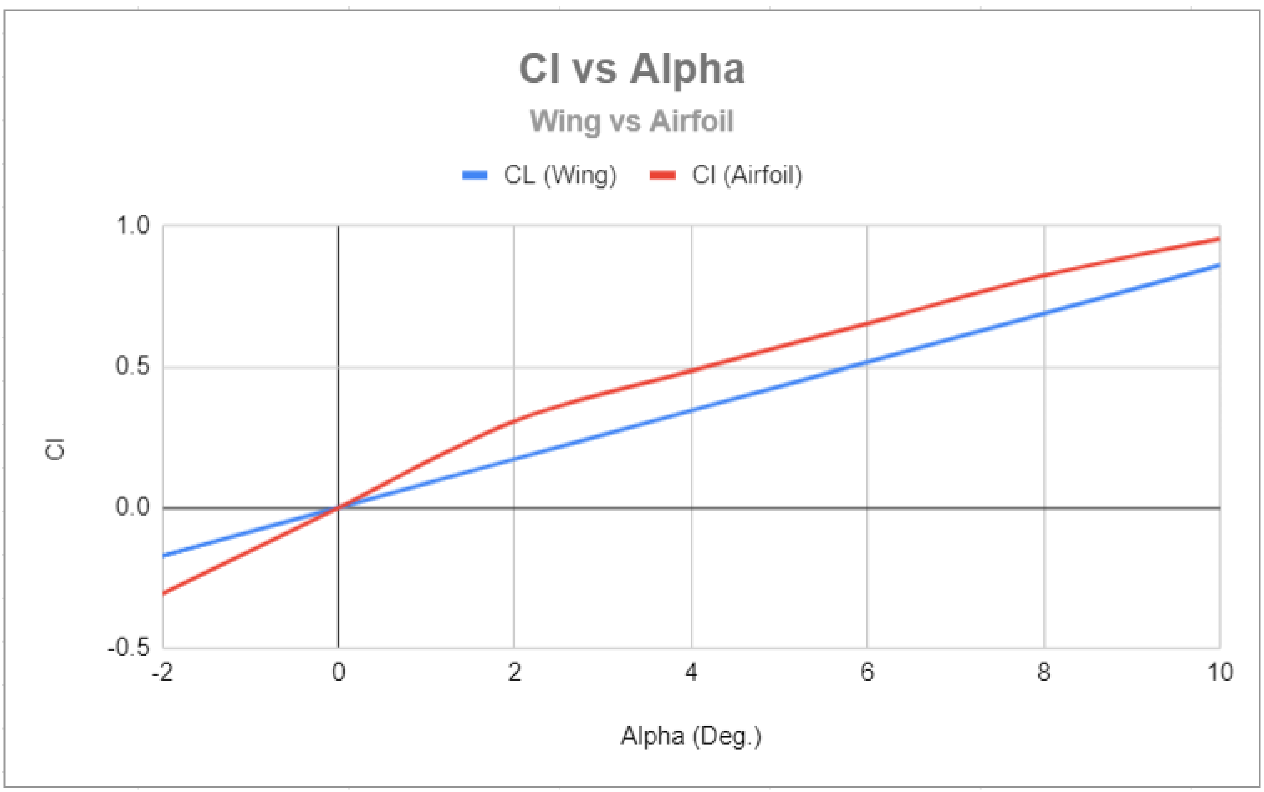


Figure 3: Cl vs Alpha plots of Wing and Airfoil

Alpha (°)	Cl (OpenVSP)	Cl (Theory) (10^{-3})	Cd (OpenVSP) (10^{-3})	Cd (Theory) (10^{-3}
-2	-0.202	-0.172	7.64	6.78
0	0	0	6.79	5.88
2	0.202	0.172	7.65	6.78
4	0.327	0.344	12.5	9.49
6	0.493	0.517	20.9	14.01
8	0.667	0.689	32.8	20.32
10	0.845	0.861	48.8	28.44

Table 2: Comparasion between OpenVSP and Lifting Line Theory Results

Wing Model

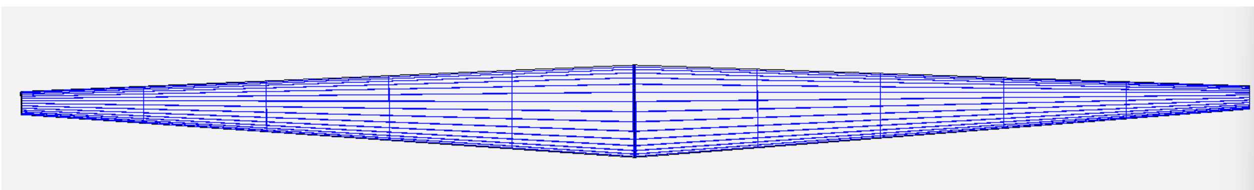


Figure 4: Top View

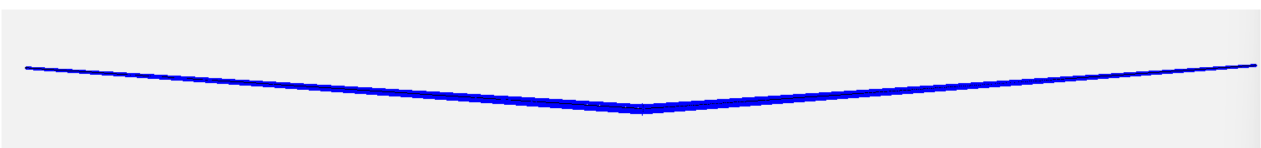


Figure 5: Front View

Observation & Inferences

- Cl vs Alpha plots obtained via Lifting Line Theory and via OpenVSP simulation results are nearly identical.
- The Coefficient of Drag of calculated using the OpenVSP simulation results is a bit more than that calculated using Drag Estimation Method. The results are nearly identical at low angles of attacks but starts deviating at higher angles of attacks.
- The Coefficient of Lift of Wing is less than that of Airfoil used. This phenomenon can be explained via formation of wingtip vertices.
- Due to finite length of wing, the air present at higher pressure below the wing starts moving towards the top surface of the wing where the pressure is low. Due to this, a component of velocity in the upward direction is induced. This changes the overall direction of the freestream velocity slightly, effectively decreasing the angle of attack. Therefore, the wing experiences lift observed at lower angles of attack even at higher angles. Hence the coefficient of lift of wing is smaller than the coefficient of lift of airfoil (or infinite wing).

Fuselage Design

Model Design

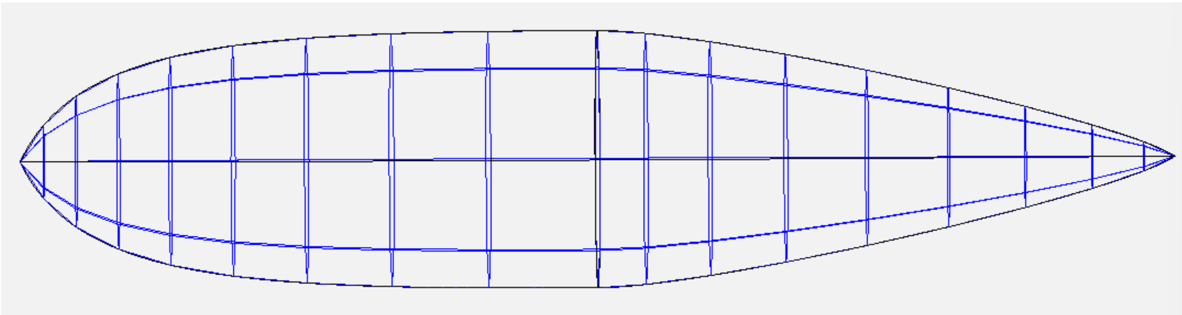


Figure 6: Wireframe View

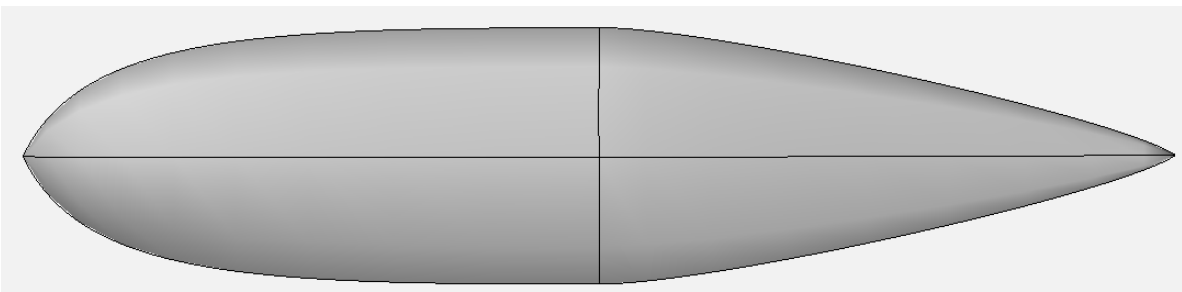


Figure 7: Shaded View

Plots

Alpha (°)	Cl (10^{-4})	Cd (10^{-4})
-2	3.91	-6.96
0	-0.42	-6.85
2	-4.83	-7.15
4	-9.31	-7.94
6	-13.7	-9.30
8	-18.0	-11.2
10	-22.0	-13.7

Table 3: OpenVSP Fuselage Simulation

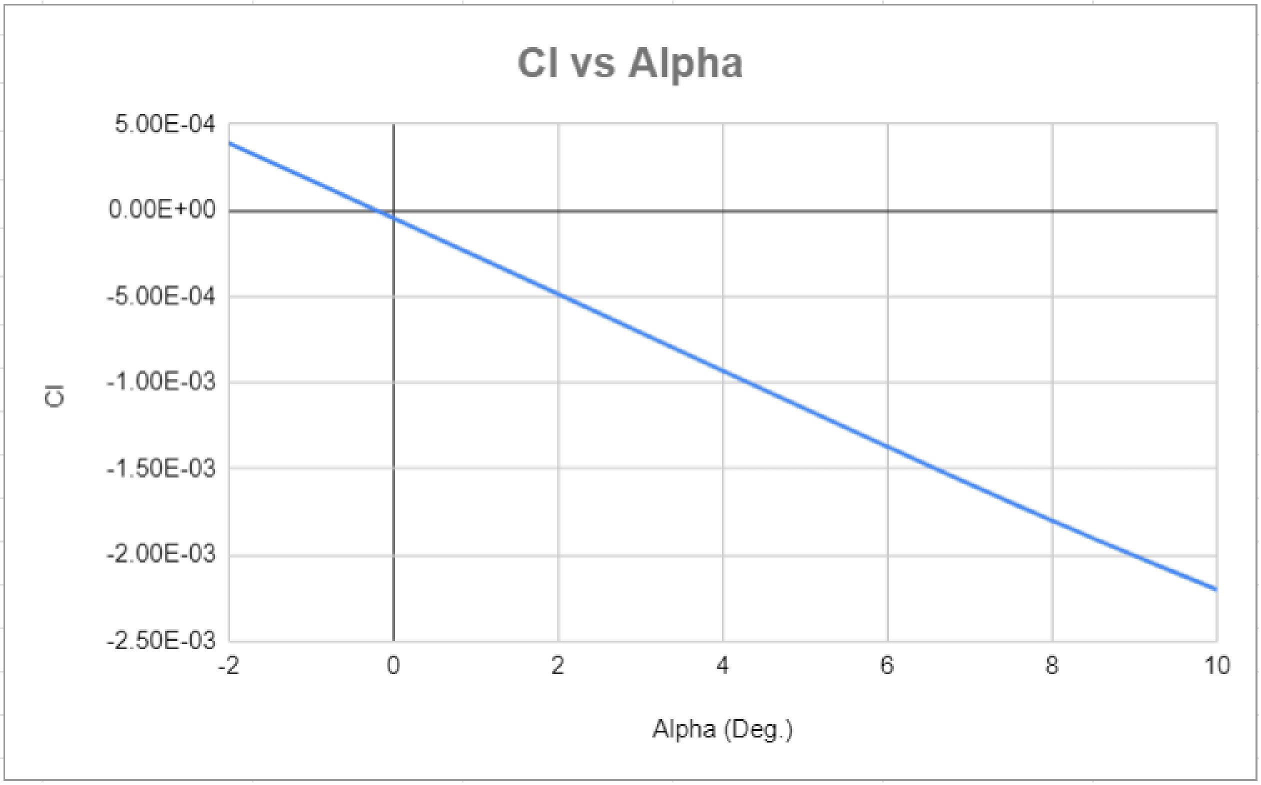


Figure 8: Cl vs Alpha Plot of Fuselage

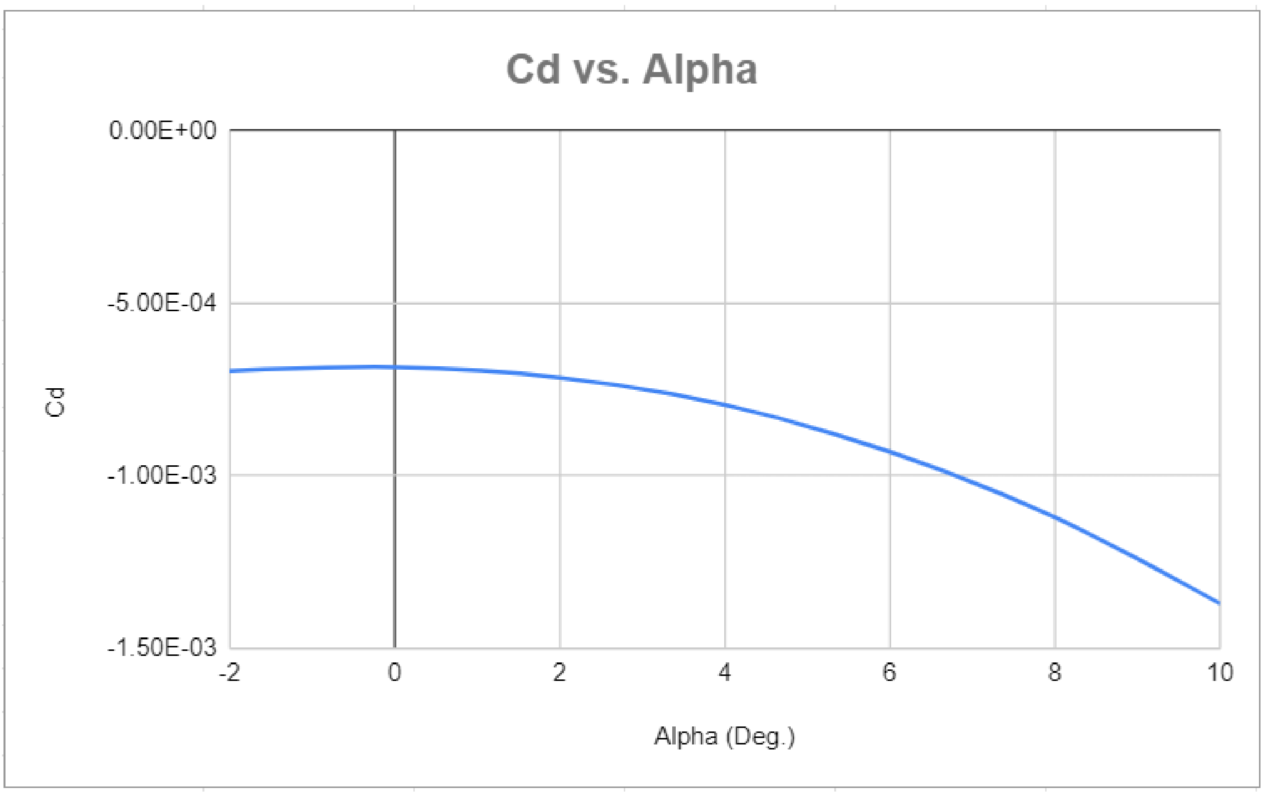
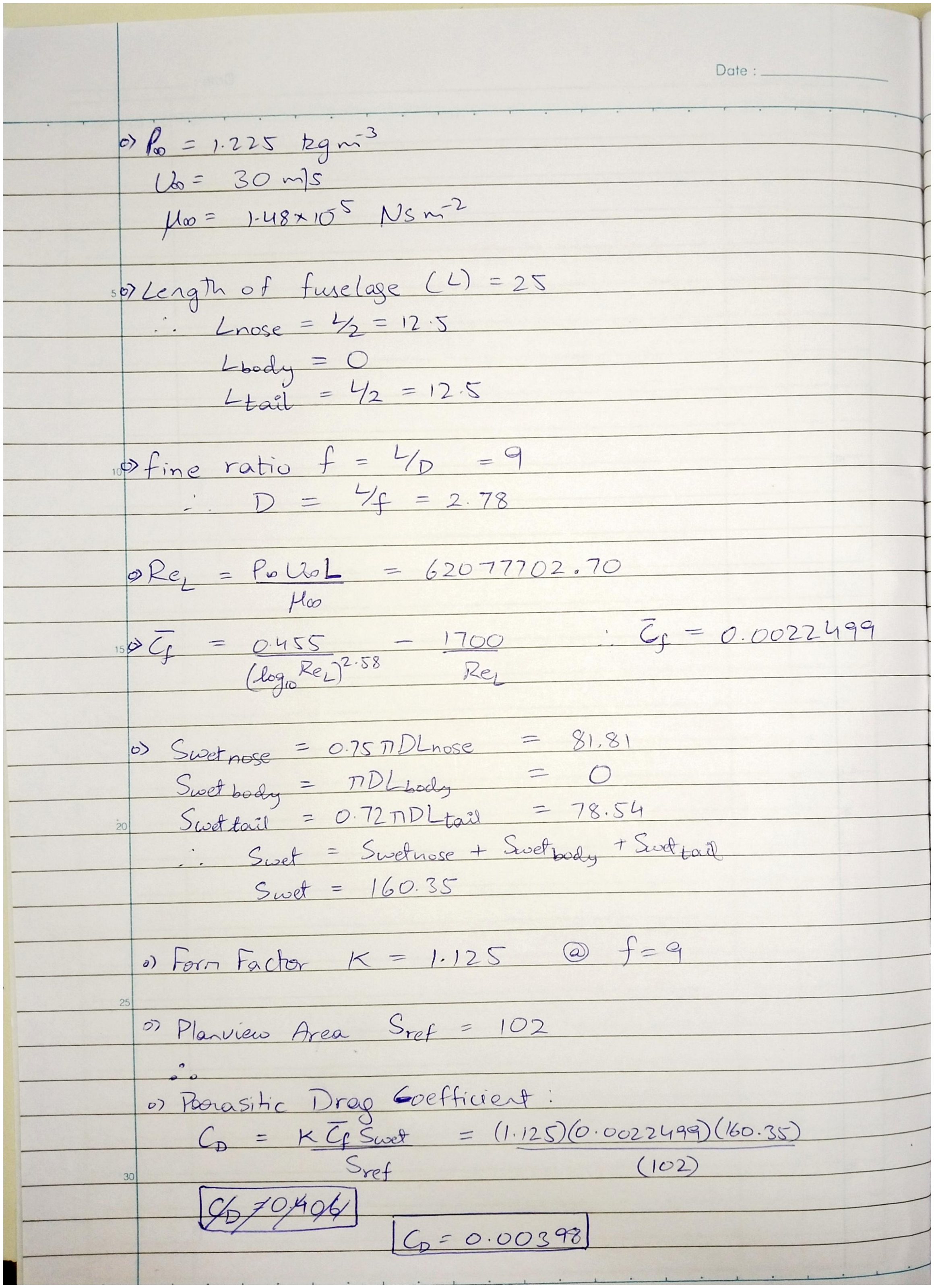


Figure 9: Cd vs Alpha plot of Fuselage

Parasitic Drag Estimation via Empirical Method



$\rho_\infty = 1.225 \ kg\,m^{-3}$

$U_\infty = 30 \ m/s$

$\mu_\infty = 1.48 \times 10^{5} \ N s\, m^{-2}$

Length of fuselage $(L) = 25$

$\therefore \ L_{nose} = L/2 = 12.5$

$L_{body} = 0$

$L_{tail} = L/2 = 12.5$

fine ratio $f = L/D = 9$

$\therefore \ D = L/f = 2.78$

$$Re_L = \frac{\rho_\infty U_\infty L}{\mu_\infty} = 62077702.70$$

$$\bar{C}_f = \frac{0.455}{(\log_{10} Re_L)^{2.58}} - \frac{1700}{Re_L} \quad \therefore \ \bar{C}_f = 0.0022499$$

$S_{wet\,nose} = 0.75 \pi D L_{nose} = 81.81$

$S_{wet\,body} = \pi D L_{body} = 0$

$S_{wet\,tail} = 0.72 \pi D L_{tail} = 78.54$

$\therefore \ S_{wet} = S_{wet\,nose} + S_{wet\,body} + S_{wet\,tail}$

$S_{wet} = 160.35$

Form Factor $K = 1.125$ @ $f = 9$

Planview Area $S_{ref} = 102$

$\therefore$

Parasitic Drag Coefficient :

$$C_D = \frac{K \bar{C}_f S_{wet}}{S_{ref}} = \frac{(1.125)(0.0022499)(160.35)}{(102)}$$

$\cancel{\boxed{C_D = 0.406}}$

$\boxed{C_D = 0.00398}$

Figure 10: Parasitic Drag via Empirical Method

Simulation Results vs Theoretical Estimate of parasitic drag

$$C_{\text{d, Estimate}} = 0.00398$$

$$C_{\text{d, OpenVSP}} == -0.000685$$

- The results obtained from OpenVSP are clearly inaccurate since the coefficient of drag is coming negative which clearly violates the second law of thermodynamics.
- If we only compared the magnitudes, still the estimated drag is roughly 5 times that of drag results from simulation.
- These results seems to appear as an issue related with the software that I am unable to resolve with my current abilities.

Horizontal Stabliser Design

Parameter	Value
Airfoil	NACA 0010
Taper Ratio	0.4
Angle of Attack	0^o
Twist Angle	0^o
Sweep Angle	10^o
Dihedral Angle	0^o
Span	19.10
Aspect Ratio	10.91

Table 4: Horizontal Stabliser Parameters

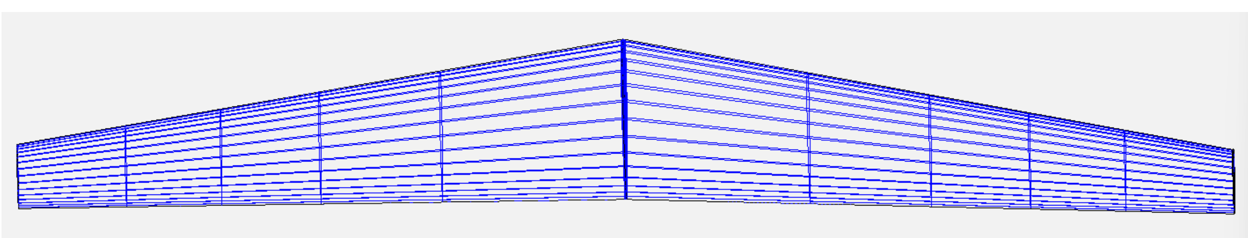


Figure 11: Horizontal Stabliser

Observation & Inferences

- The Cl vs Alpha curve is a straight line passing through origin having slope 0.088 rad^{-1}.
- The Cd vs Alpha curve is upwards parabolic with $\text{Cd}_{\text{min}} = 0.00682$ at alpha $= 0^o$.

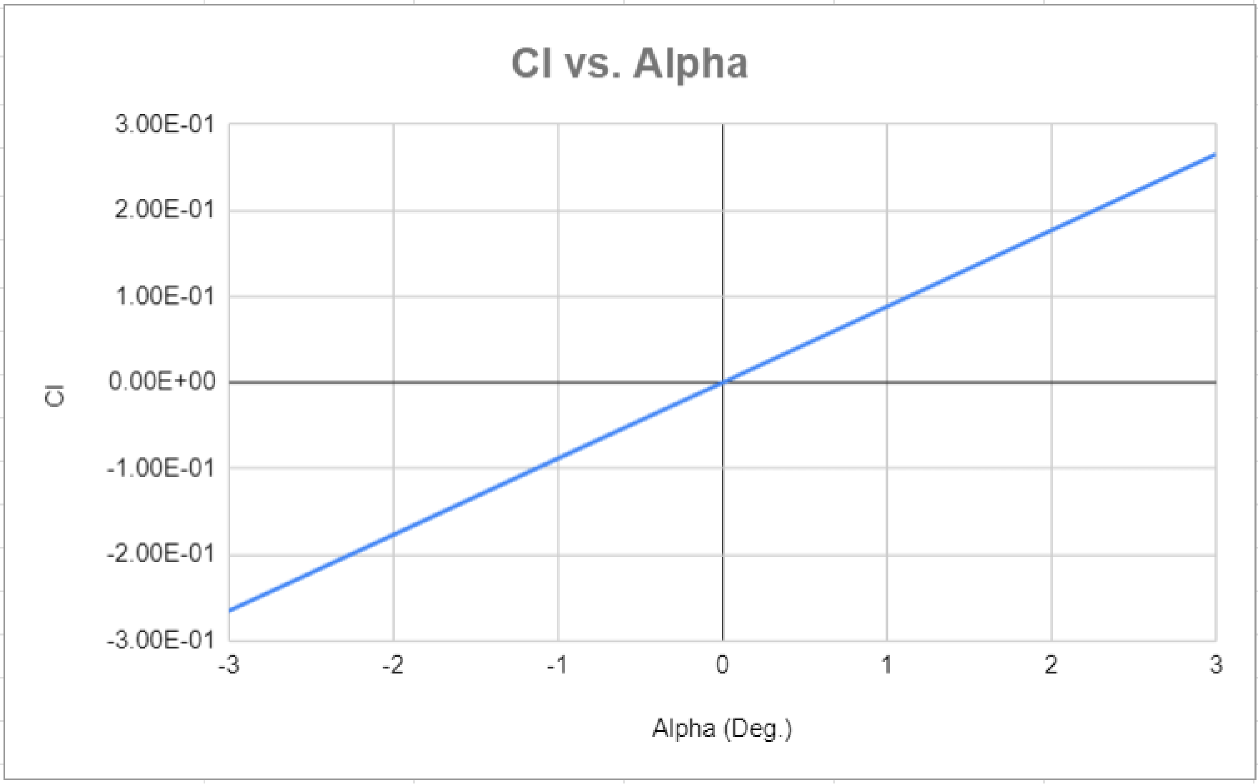


Figure 12: Cl vs Alpha

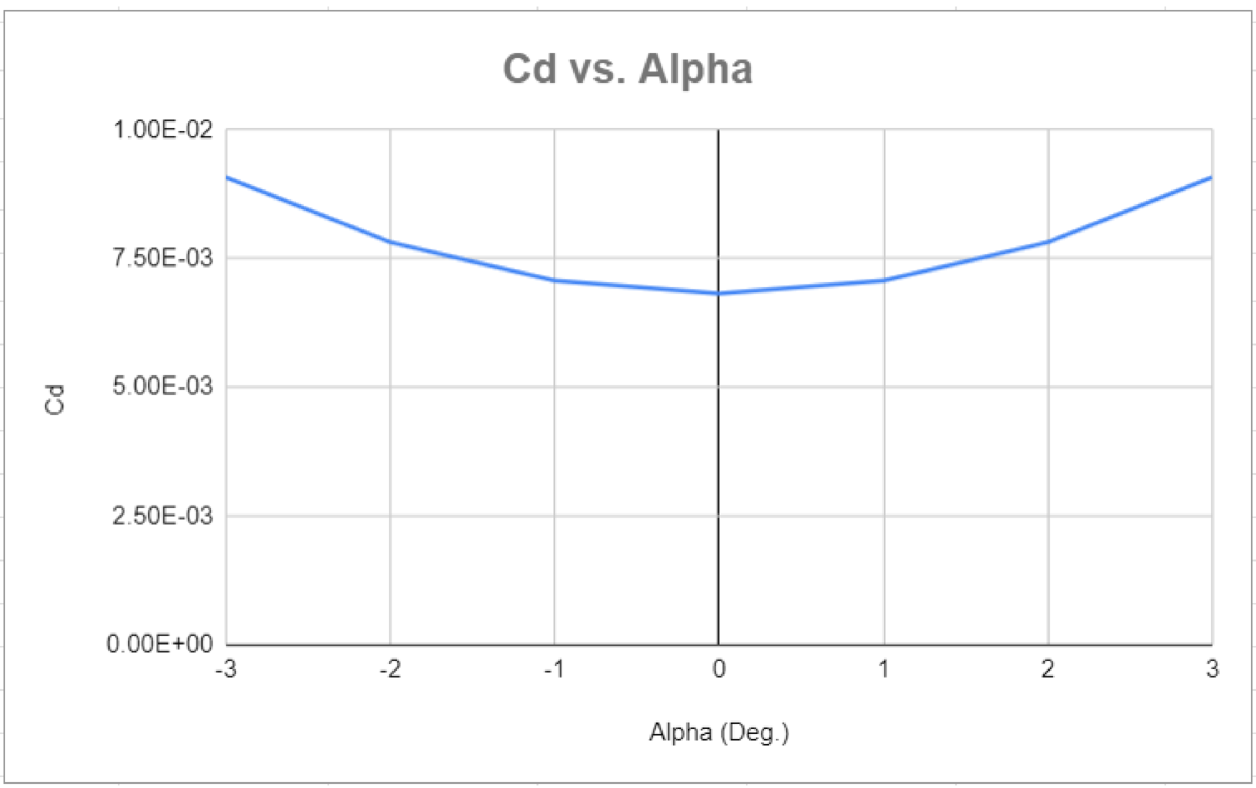


Figure 13: Cd vs Alpha

Vertical Stabliser Design

Model Parameters

Parameter	Value
Airfoil	NACA 0010
Taper Ratio	0.4
Angle of Attack	0^o
Twist Angle	0^o
Sweep Angle	30^o
Dihedral Angle	0^o
Span	3.8
Aspect Ratio	2.17

Table 5: Vertical Stabliser Parameters

Model Design

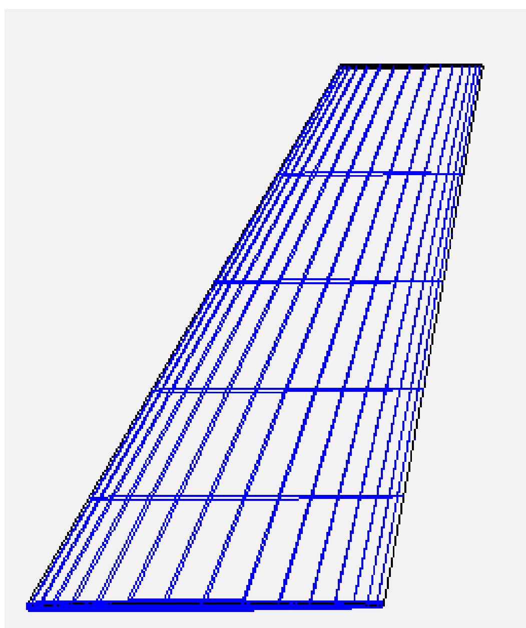


Figure 14: Vertical Stabliser

Observation & Inferences

- Cl = 0 at all alpha. It makes sense cause vertical stabliser is not supposed to create lift as air passes laterally about it.
- Cd vs Beta plot is parabolic with $Cd_{min} = 0.00136$ at Beta = 0^o.
- CFy is the force in Y-direction, if freestream velocity is in X-direction and lift is produced in the Z-direction. It varies linearly with Beta, passing through origin and having slope as -0.00973 rad^{-1}.

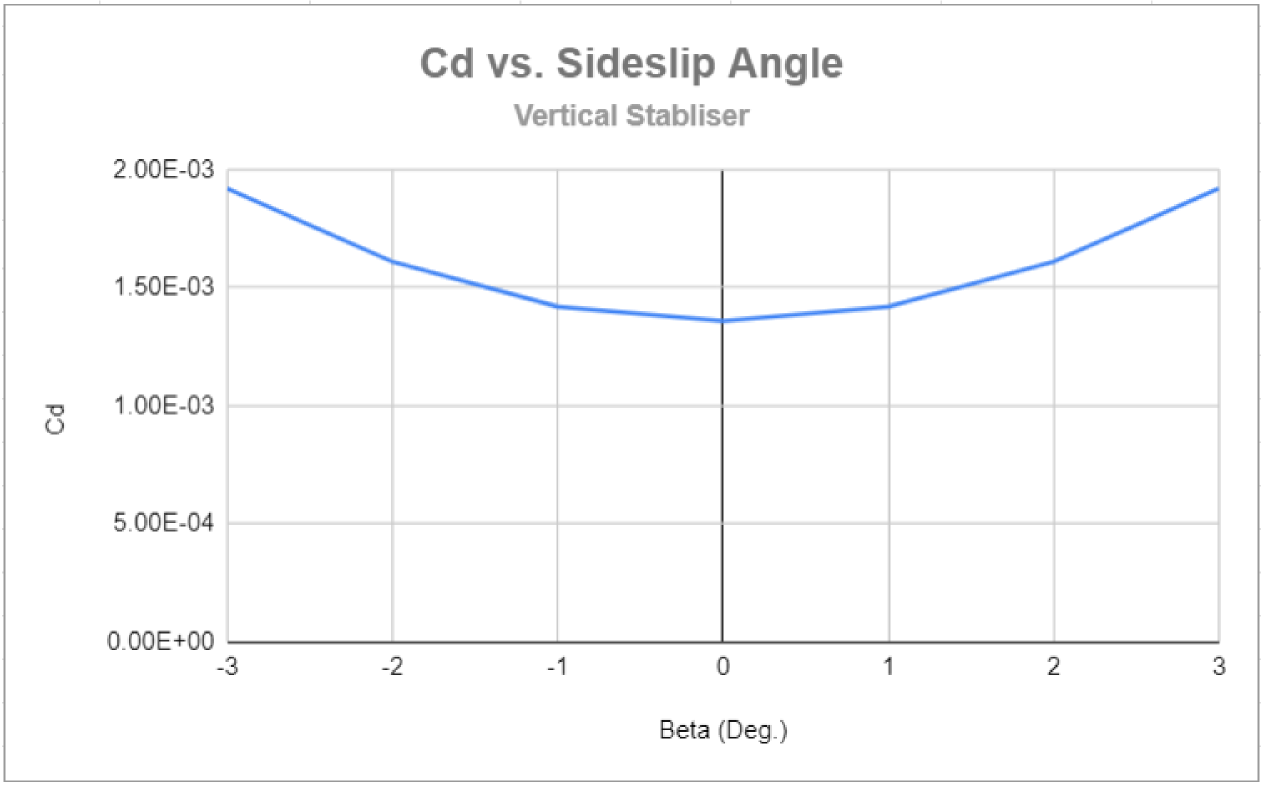


Figure 15: Cd vs Sideslip angle

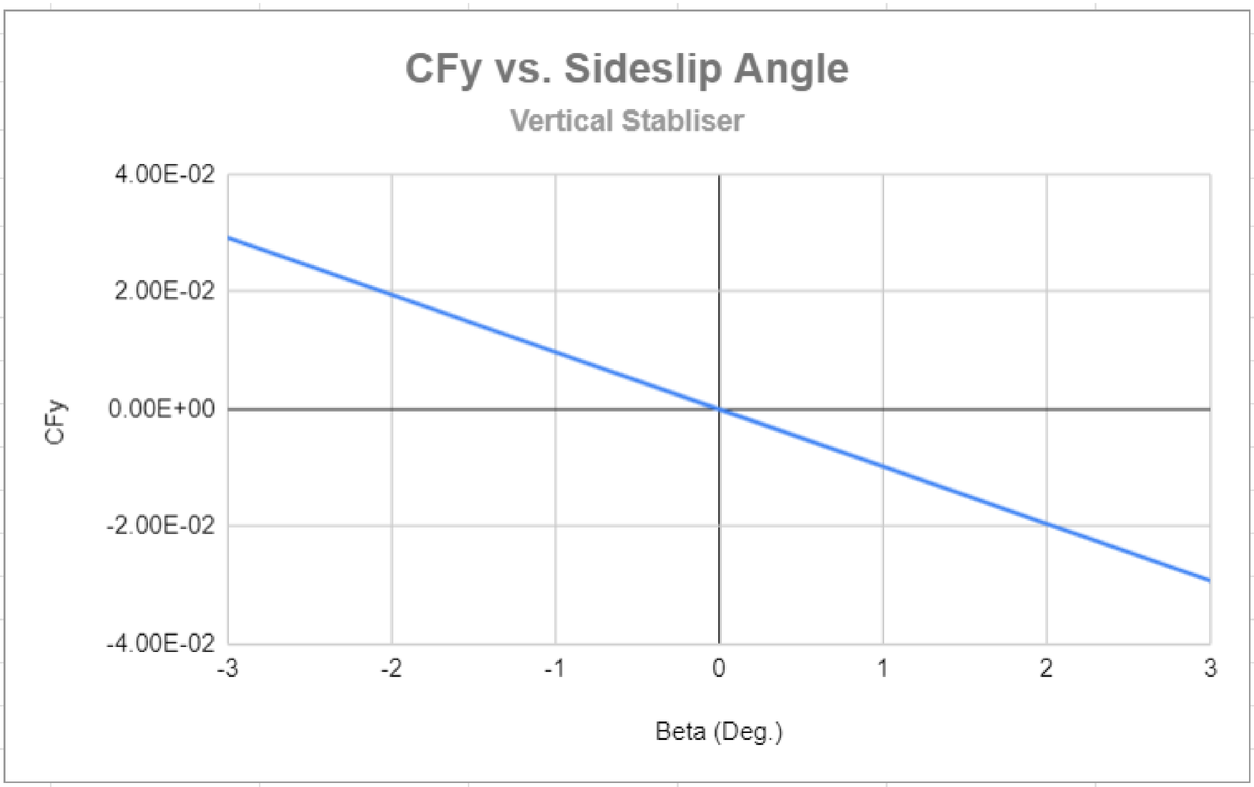


Figure 16: CFy vs Sideslip angle

Glider

Model Design

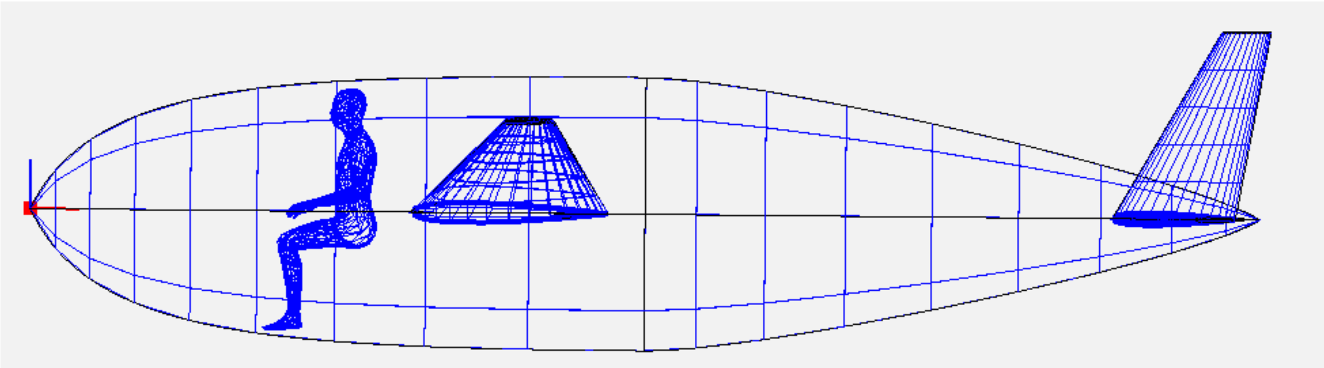


Figure 17: Side View

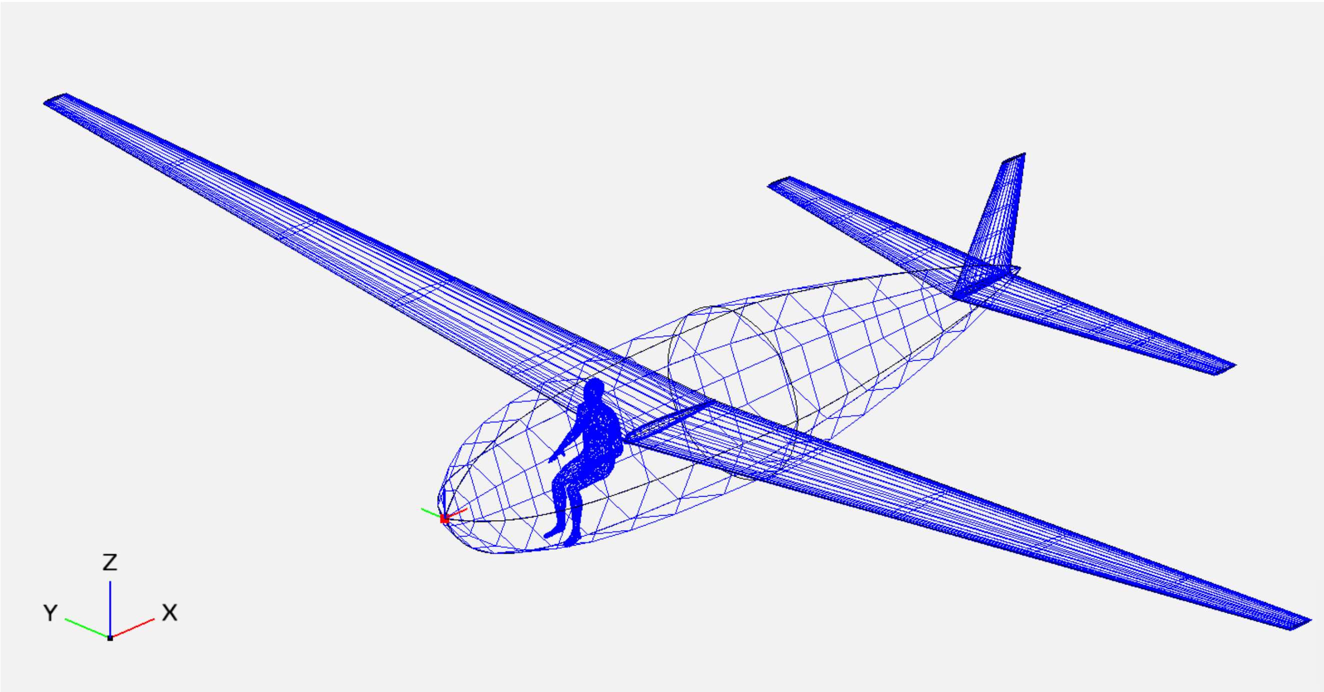


Figure 18: Tilted View

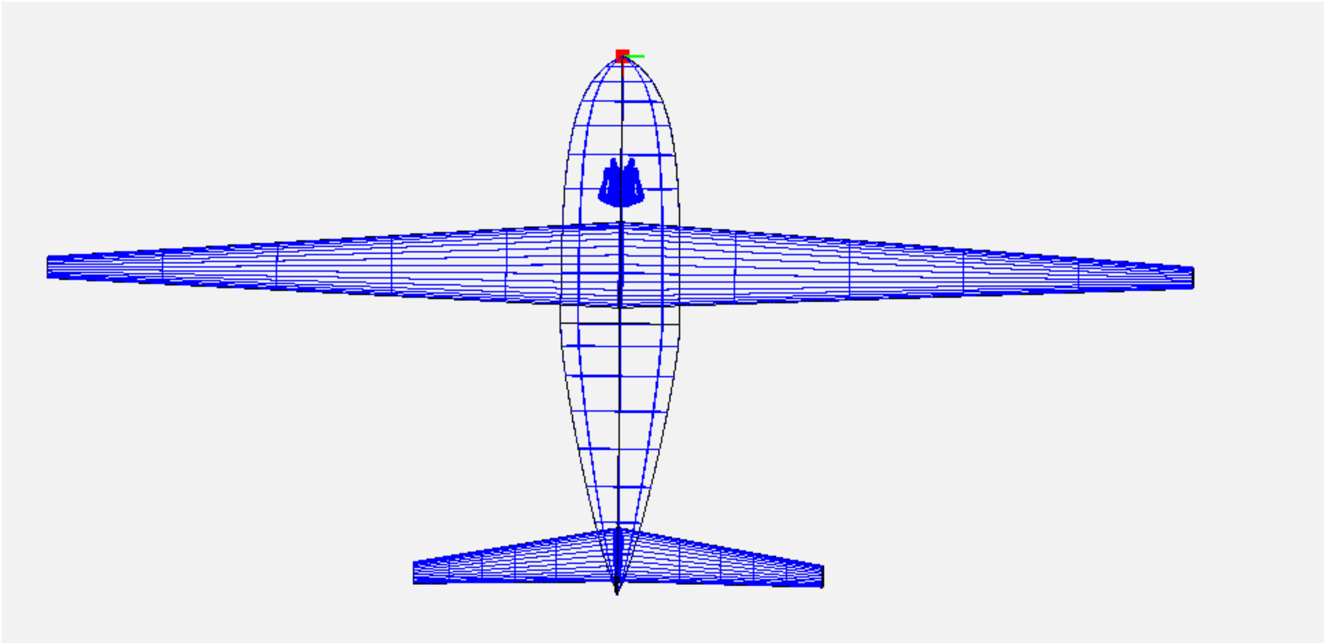


Figure 19: Top View

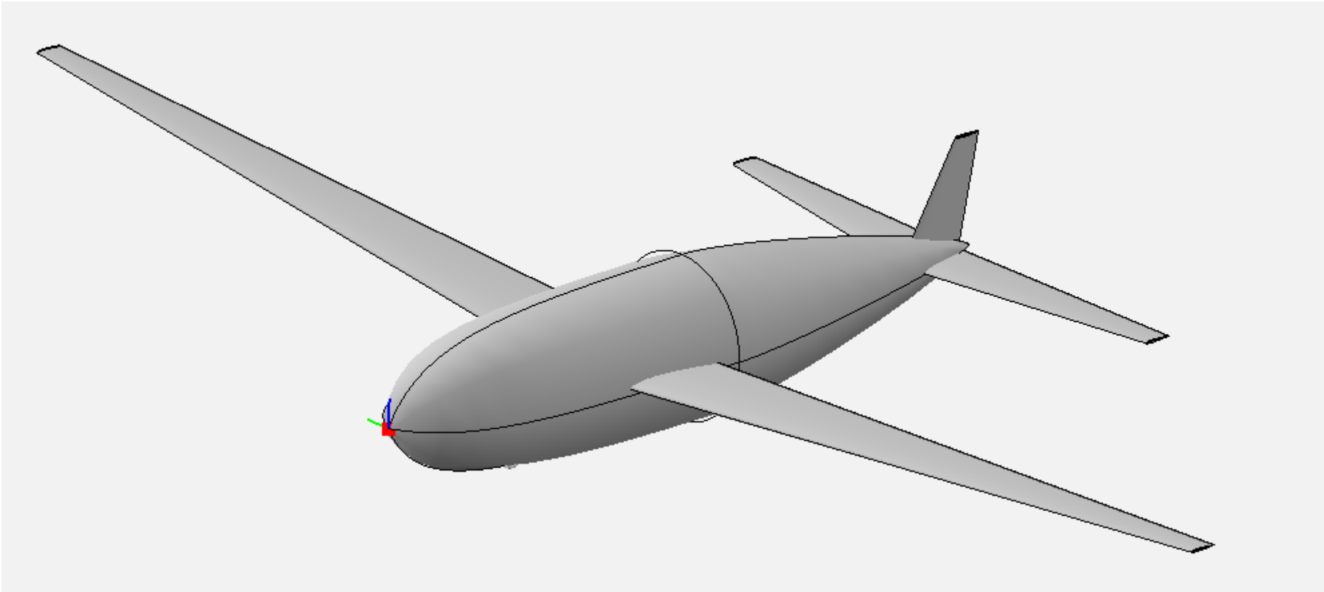


Figure 20: Glider Model

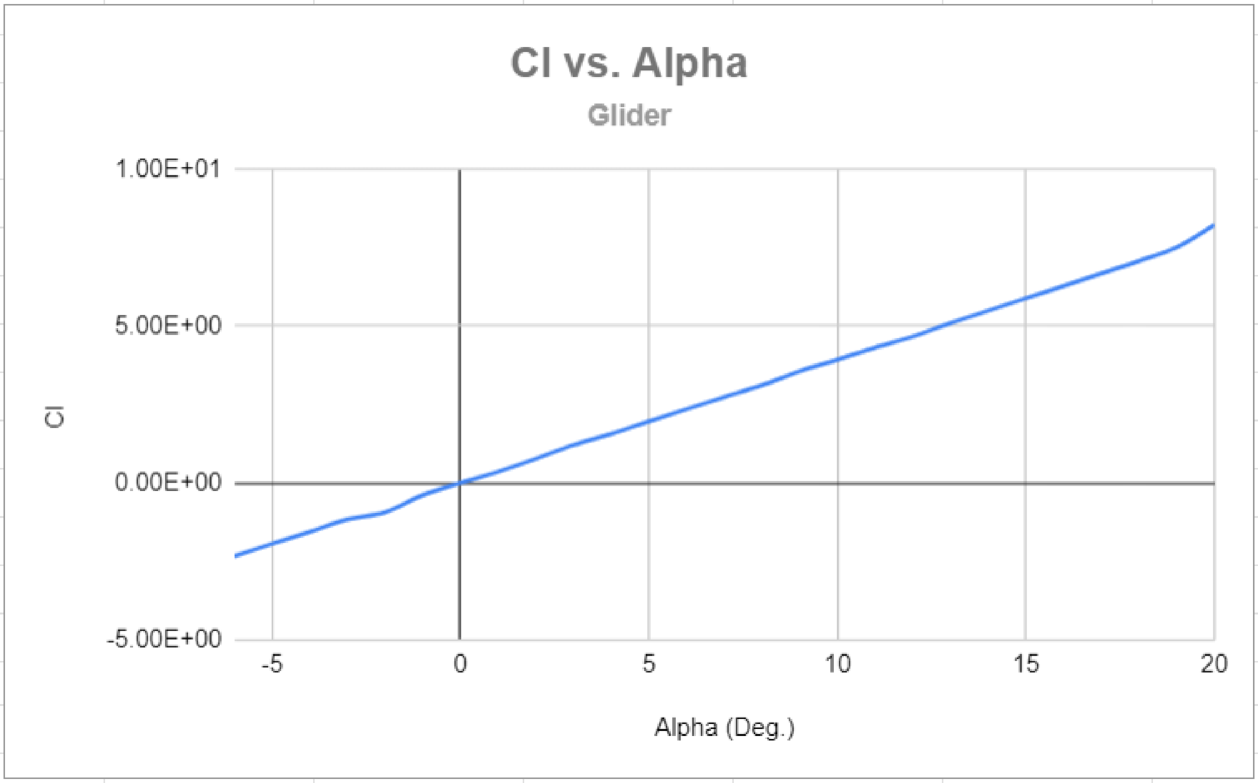


Figure 21: Cl vs Alpha Plot

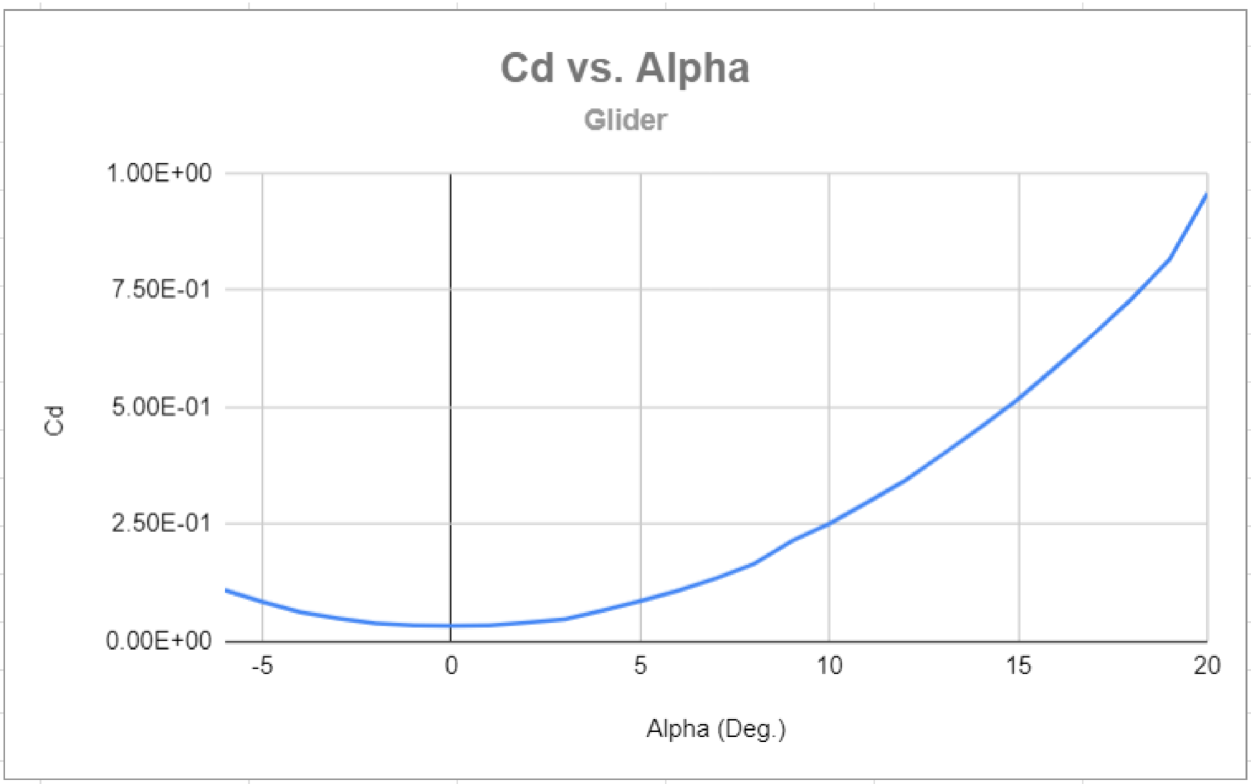


Figure 22: Cd vs Alpha Plot

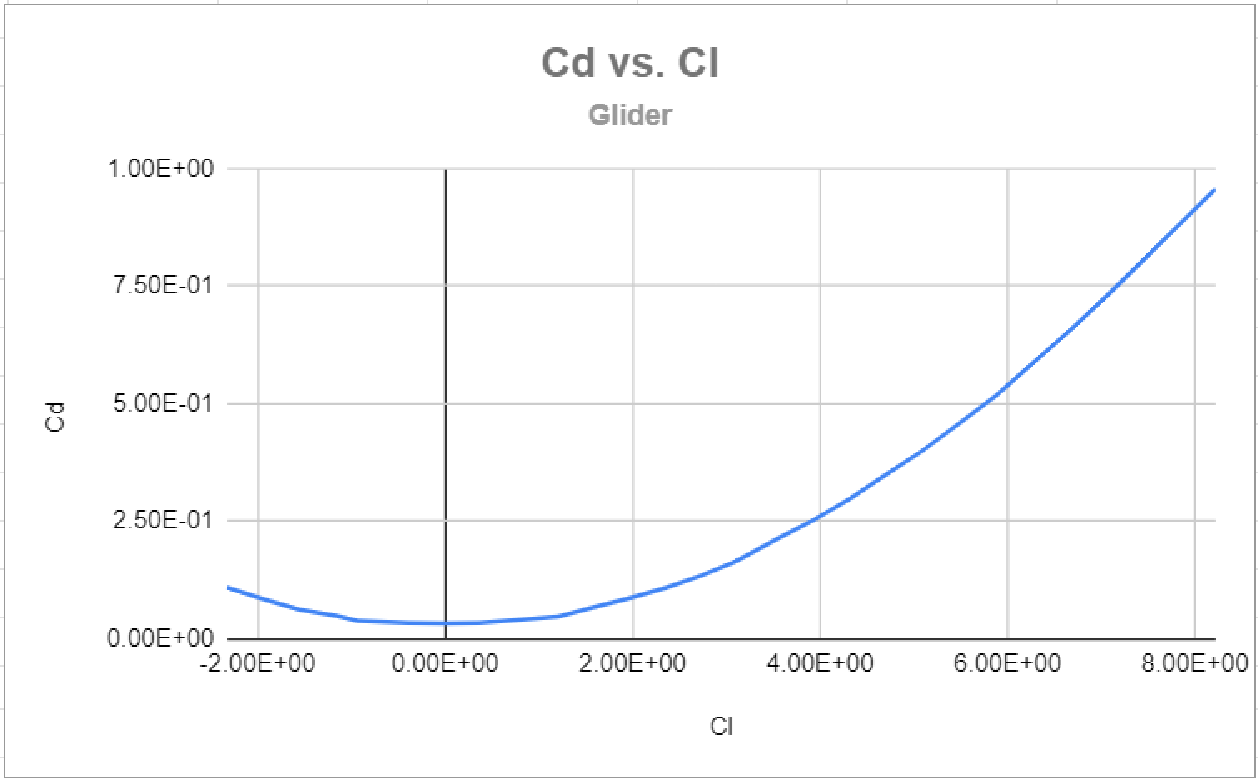


Figure 23: Cd vs Cl Plot

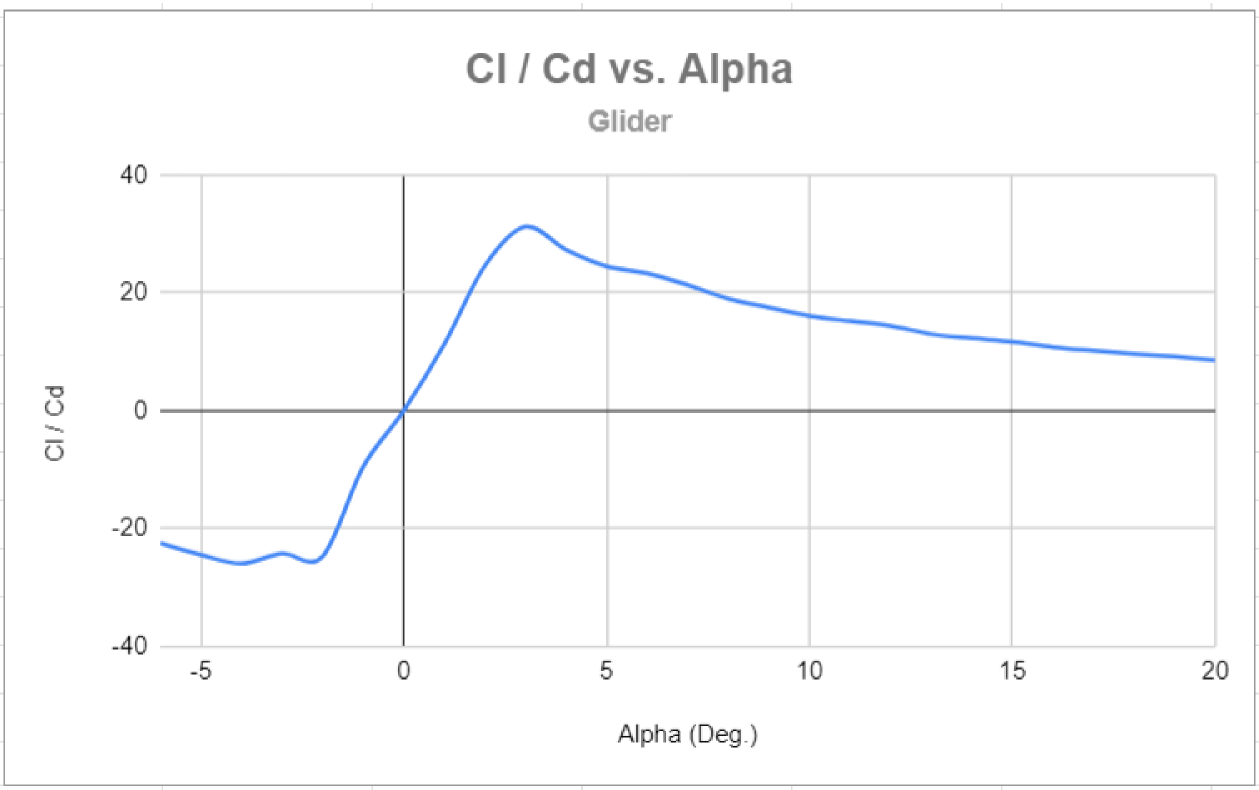
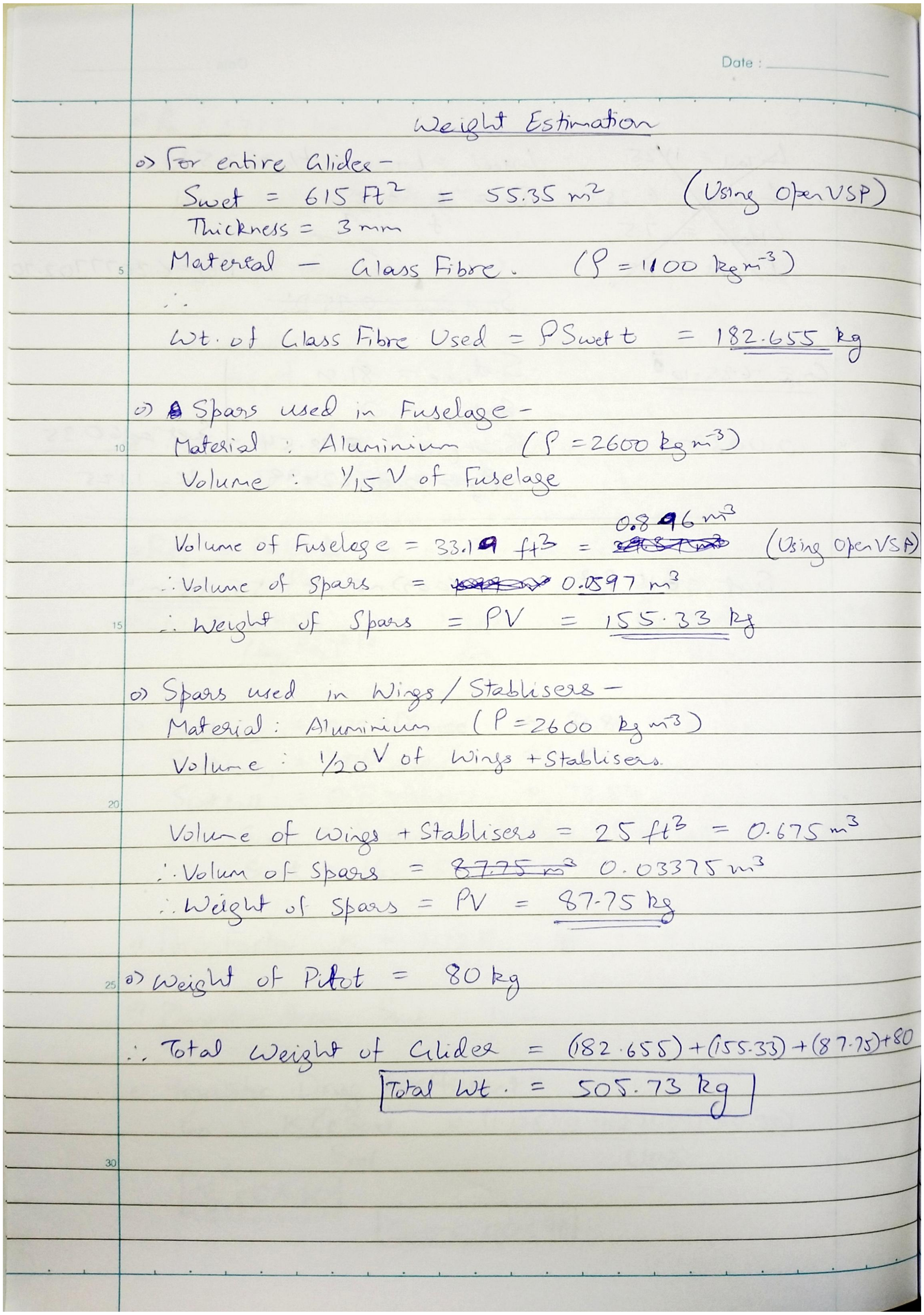


Figure 24: Cl/Cd vs Alpha Plot

Weight Estimation of Glider



Date :

Weight Estimation

o) For entire Glider –

S_{wet} = 615 ft^2 = 55.35 m^2 (Using OpenVSP)

Thickness = 3 mm

Material – Glass Fibre. (ρ = 1100 kg m^{-3})

$\therefore$

Wt. of Glass Fibre Used = $\rho S_{wet} t$ = 182.655 kg

o) Spars used in Fuselage –

Material : Aluminium (ρ = 2600 kg m^{-3})

Volume : 1/15 V of Fuselage

Volume of Fuselage = 33.19 ft^3 = 0.896 m^3 (Using OpenVSP)

$\therefore$ Volume of Spars = 0.0597 m^3

$\therefore$ Weight of Spars = ρV = 155.33 kg

o) Spars used in Wings / Stablisers –

Material : Aluminium (ρ = 2600 kg m^3)

Volume : 1/20 V of Wings + Stablisers.

Volume of Wings + Stablisers = 25 ft^3 = 0.675 m^3

$\therefore$ Volum of Spars = ~~87.75 m^3~~ 0.03375 m^3

$\therefore$ Weight of Spars = ρV = 87.75 kg

o) Weight of Pilot = 80 kg

$\therefore$ Total Weight of Glider = (182.655) + (155.33) + (87.75) + 80

$\boxed{\text{Total Wt.} = 505.73 \text{ kg}}$

Figure 25: Glider Weight Calculations

Glider Weight

$$Total\ Weight\ of\ Glider\ (W) = 505.73kg$$

Optimal Glide Angle

$$Glide\ Angle = \frac{1}{L/D} = \frac{1}{C_{\mathrm{l}}/C_{\mathrm{d}}}$$

For Maximum Range, Glide Angle should be minimum. Hence, Cl/Cd ratio should be maximum. Cl/Cd is maximum at Alpha = 3^{o} with Cl/Cd $)_{\mathrm{max}} = 31.2$
Therefore,

$$Optimum\ Glide\ Angle\ (\gamma) = 1.83^{\mathrm{o}}$$

Range

$$Optimal\ Range = 31\ km$$

Glide Speed

$$W cos\gamma = L = \frac{1}{2}\rho U^2 S C_{\mathrm{l}}$$

So,

$$U = \sqrt{\frac{2W cos\gamma}{\rho S C_{\mathrm{l}}}}$$

Therefore,

$$U = 5\ m/s$$

Descent Rate

$$Descent\ Rate = U sin\gamma = 0.159\ m/s$$

Scope of Improvements in Glider Design

- A cambered airfoil, with more suitable parameters can be chosen instead of the currently used NACA 0010. The new airfoil can be of higher camber and of suitable thickness. This airfoil when used for the wing and horizontal stablisers, will develop larger lifts and hence larger L/D ratios. So the range of the glider will increase as well as the descent rate will decrease.

- Control Surfaces should be included in the glider such as flaps, elevators, rudder, ailerons, etc. This will ensure that in case of some emergency, the pilot can be able to control the glider to land safely.

- The fuselage can be improved to be smaller and be build more aerodynamically efficient.

- Although the glider is able to satisfy the design requirements, but that is only applicable when operated in optimal state. In case of disturbances, which are sure to be present in the atmosphere like gusts, birds, etc. the glider will not behave optimally, and may crash in worst case scenario. So, the glider needs to be redesigned to cater to the presence of such disturbances and fly safely.
- The results provided by OpenVSP are not entirely accurate. They involve alot of approximations and does not account for alot of the involved parameters such as viscosity, etc. in order to achieve faster calculations. Using Ansys simulation results can provide much better data with much better accuracy and will be more practically applicable during the prototype testing phase.

Acknowledgement

Alot of my friends helped me during my assignment. Some key mentions include Rohan Chowdhury and Binay Kumar Shaw, Chaitanya Keshri and Devesh Mittal. They cleared my doubts throughout the assignment and made my concepts stronger. It would have been a lot more difficult to complete the assignment without their sincere help.
Our course instructor, Dhwanil sir also helped alot. His lectures made my theoretical concepts stronger.

References

- Lecture Slides
- Latex Documentation and help sites
- wikipedia page on airfoils $https://en.wikipedia.org/wiki/NACA_airfoil$
- Youtube videos on Lifting Line Theory
- OpenVSP tutorials on youtube