AE244

LOW SPEED AERODYNAMICS

ASSIGNMENT-3

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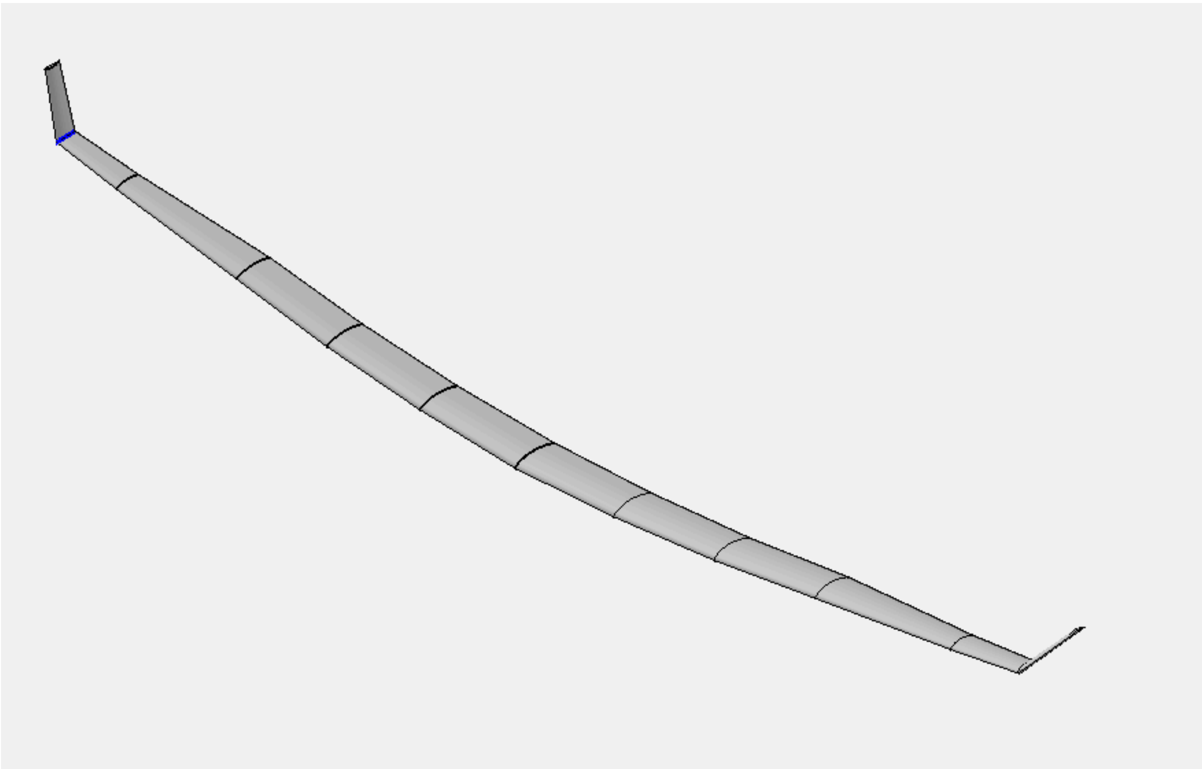
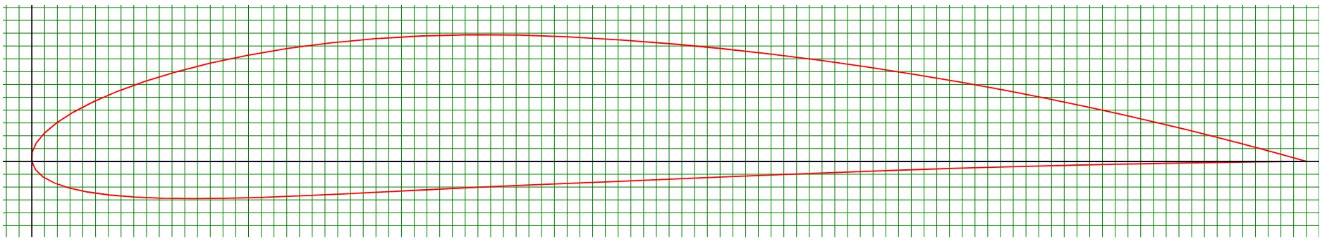
1. Wing Design

1.1 Description of Wing -

Airfoil	NACA 4412
Span	28 m
Dihedral	3°
Twist	0
Sweep	0
Angle of Attack	0
Taper	0.347
Number of section in semi wing	6
Root chord	2.082 m
Tip chord	0.723 m
Attached Devices	Winglets



Why NACA 4412 ?



Name = NACA 4412 Airfoil M=4.0% P=40.0% T=12.0%
Chord = 1000mm Radius = 0mm Thickness = 100% Origin = 0% Pitch = 0°

I choose NACA 4412 Airfoil for my glider because of these listed points-

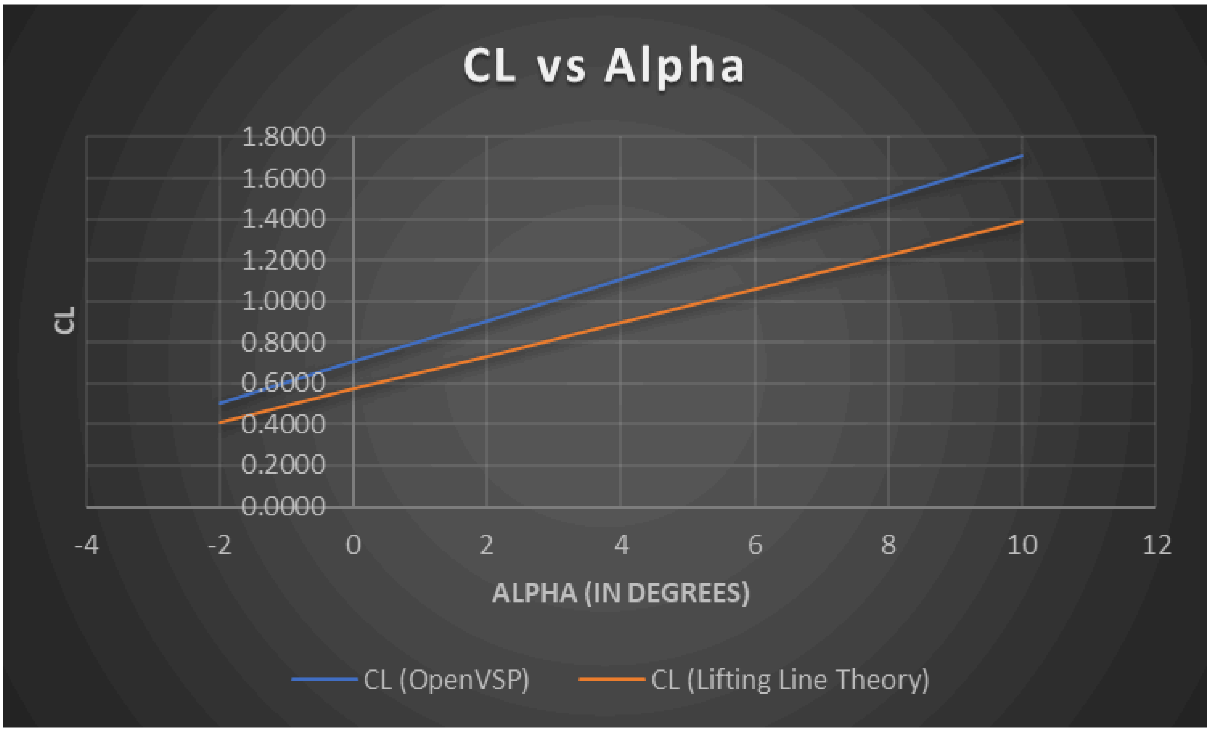
1. Performance: The NACA 4412 airfoil is known for its favourable lift-to-drag ratio, which is crucial for glider performance.

2. Historical Performance: The NACA 4412 has a history of successful use in various aircraft designs, including gliders. Examples -
    1. Aeronca 65-TAC Defender
    2. Aeronca 11-AC Chief

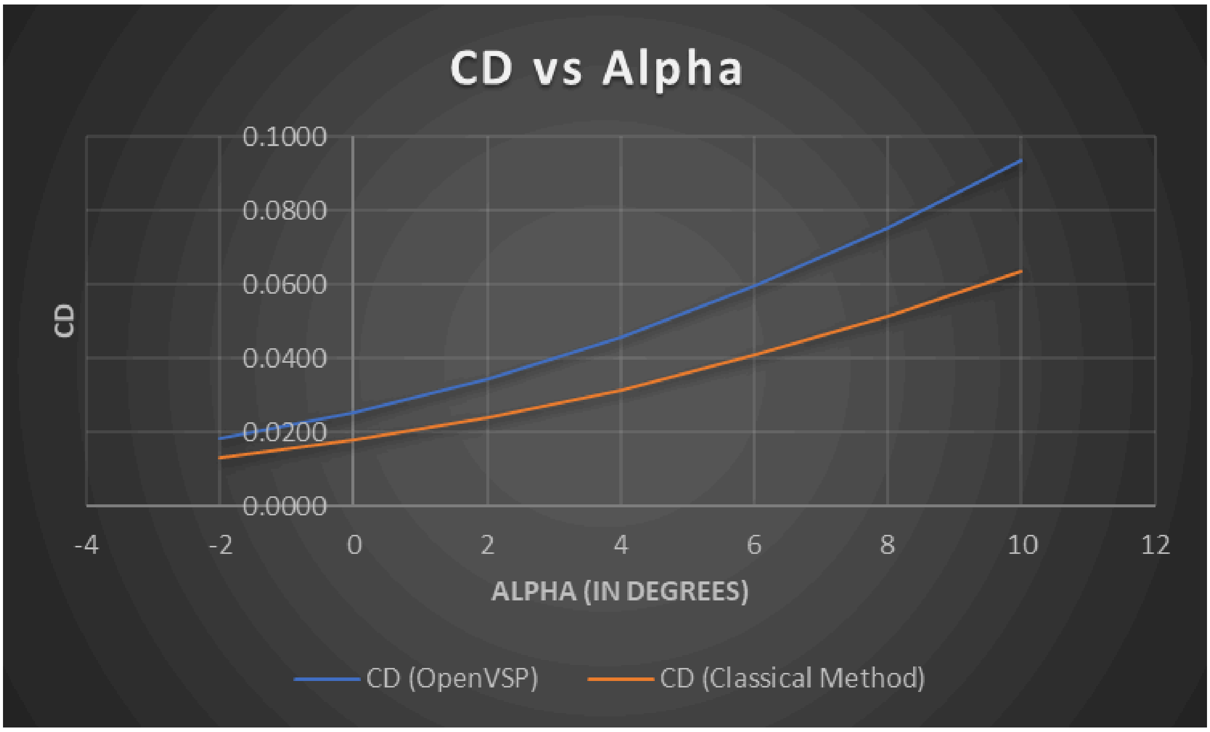
3. Stall Characteristics: It tends to provide a more predictable and manageable behaviour near its stall angle compared to some other airfoils.

4. Versatility: The NACA 4412 is a versatile airfoil that performs well across a range of angles of attack and Reynolds numbers.

1.2 CL vs Alpha (Lifting Line Theory and OpenVSP) (α = -2° to 10°)

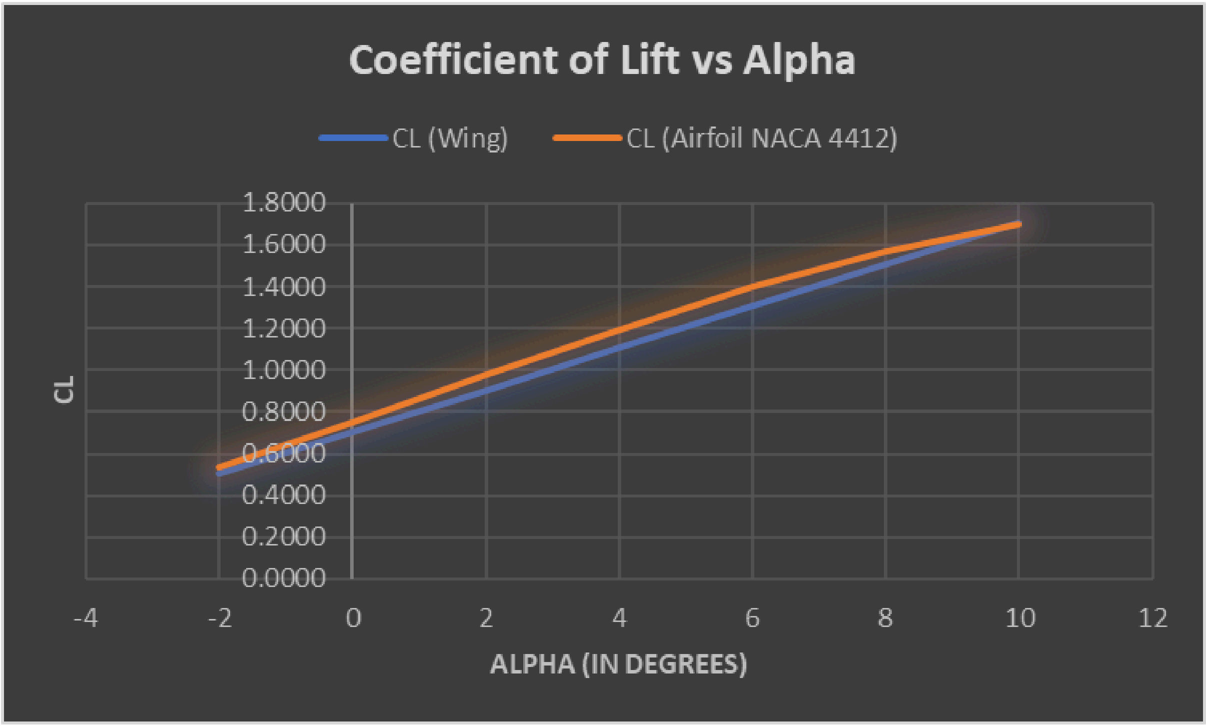


1.3 CD vs Alpha (Classical Method and OpenVSP)



Parasitic Drag Coefficient = 0.00845

1.4 CL vs Alpha (Wing and Airfoil)



1.5 Observations and Interpretations

While comparing the lifting line theory and VSP calculations, we notice only minor differences, with the lifting line theory encompassing all parameters captured by VSP.

Assumptions in LIfting Line theory:

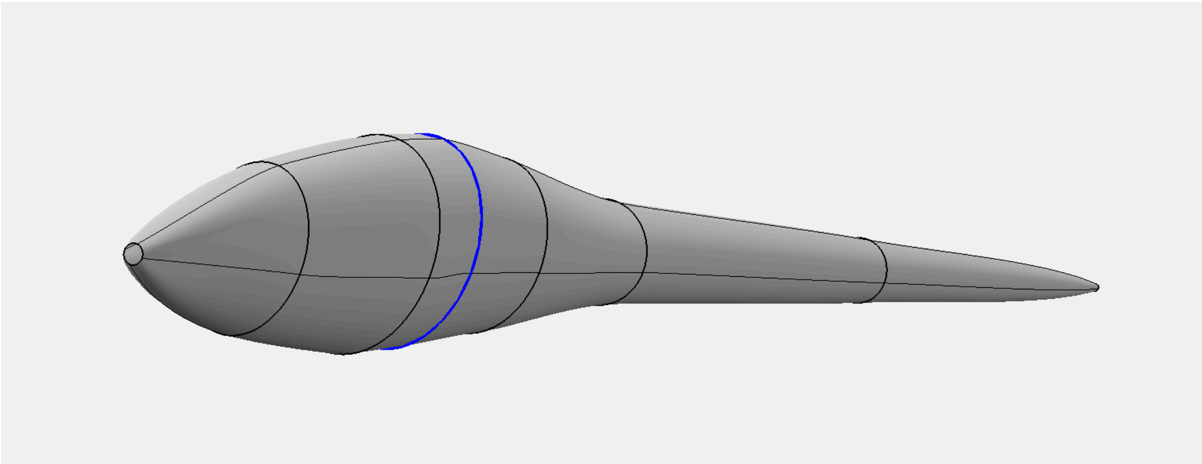
1. Inviscid Flow
2. 2D WIng
3. Thin Airfoil
4. Steady and Incompressible Flow

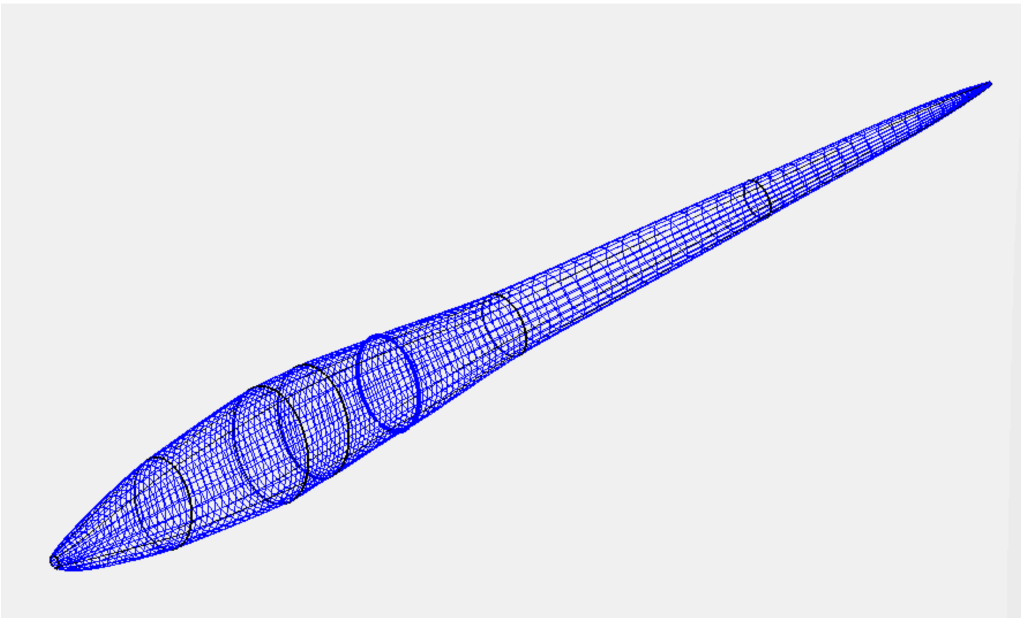
These are assumptions taken in Lifting Line Theory , so these will somehow affect actual values.

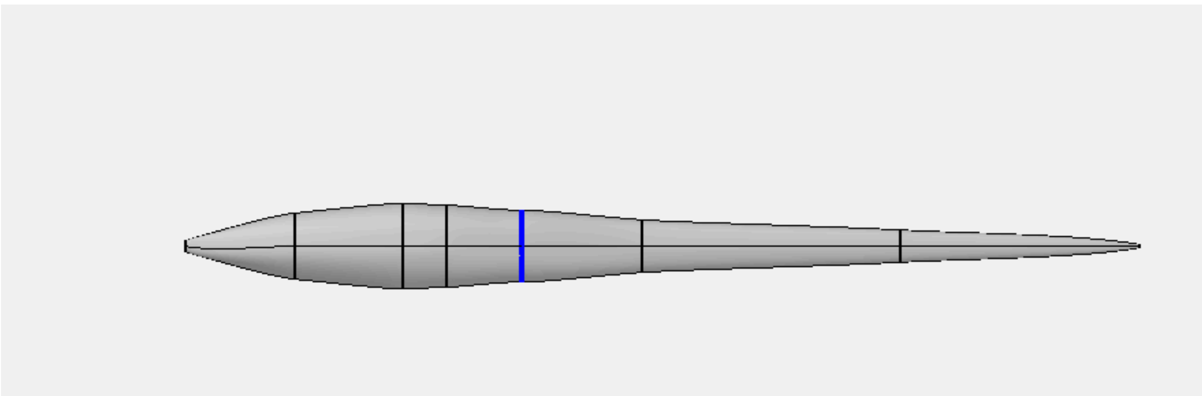
Also here in OpenVSP, it does not show stalls. As I see CL keeps on increasing if we increase angle of attack.

2. Fuselage Design

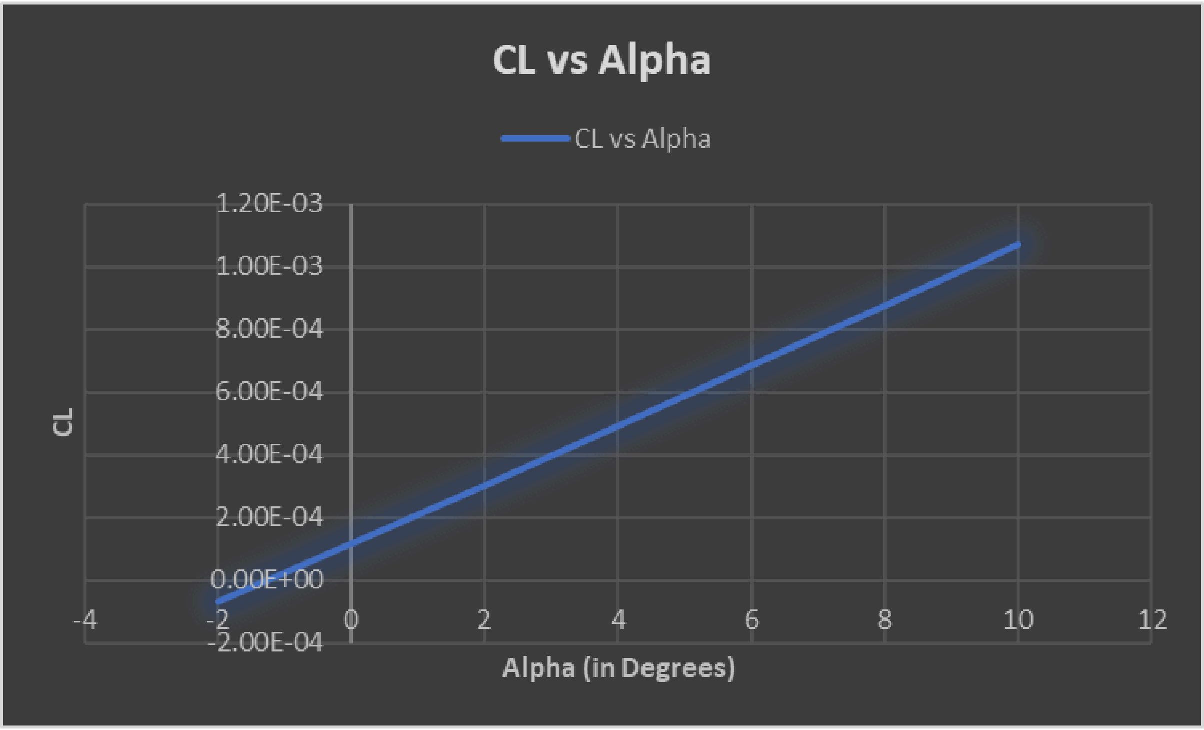
2.1 CAD of the fuselage

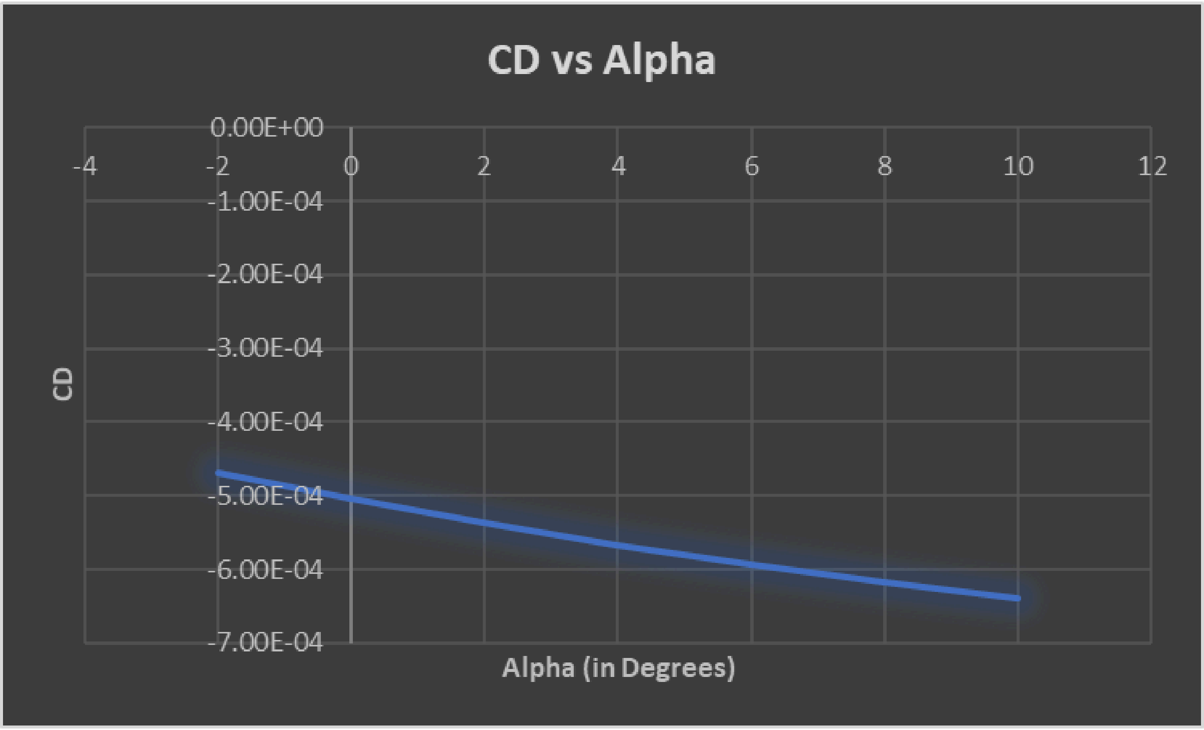






2.2 Lift and Drag Estimates From OpenVSP





2.3 Parasitic drag estimates based on empirical method

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

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

$$C_{D_0} = \sum_{i=1}^{N} \frac{K_i \bar{C}_{f_i} S_{\mathrm{wet}_i}}{S_{\mathrm{ref}}}$$

Re_L = (1.225 * 30 * 15.24) / 1.5 * 10^{-5} = 3.73 * 10^{7}

C_f = (0.455/ ($\log_{10}$(3.73 * 10^{7}))$^{2.58}$) - (1700 / 3.73 * 10^{7}) = 0.00240

C_{D0} = 0.00782

2.4 Comparisons and Comments

We calculate the parasitic drag coefficient using above formulas , we get C_{Do} as 0.00782 which is very close to the value obtained through the Classical method as stated in section 1.3.

The reason for this small difference is due to the assumptions made in classical method for calculating the parasitic drag coefficient.

Assumptions made in classical method -

1. Inviscid flow
2. No separation
3. Inviscid and Incompressible flow

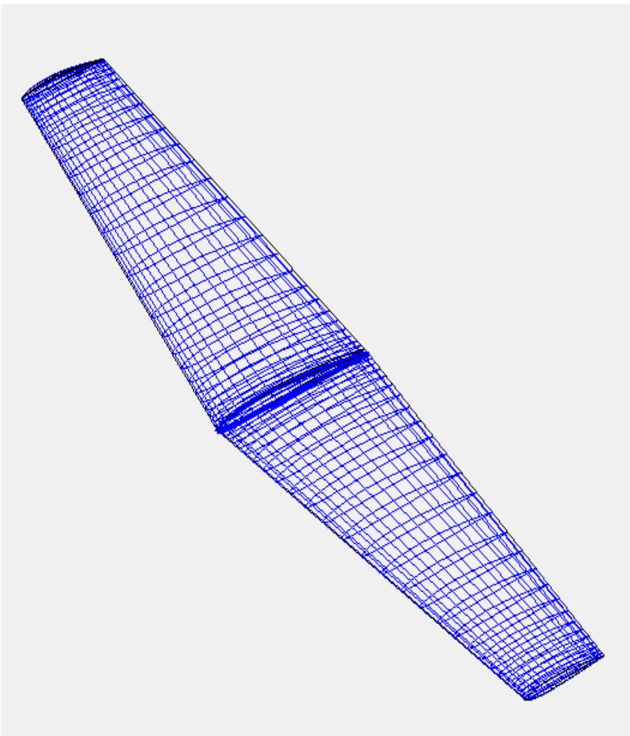
The classical method value is just 8.05 % more than the value obtained here.

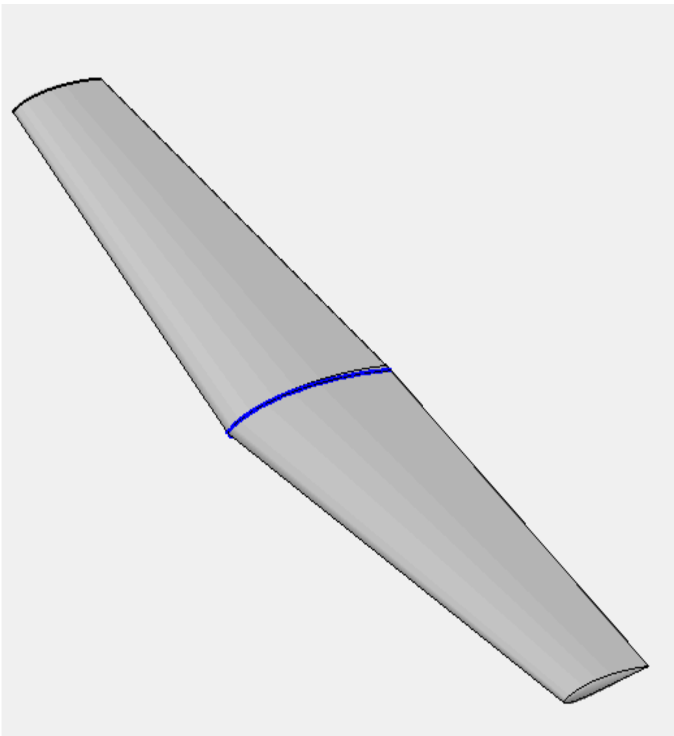
3. Stabilizer Design

3.1 Horizontal and Vertical stabilizer designs

Horizontal Stabiliser

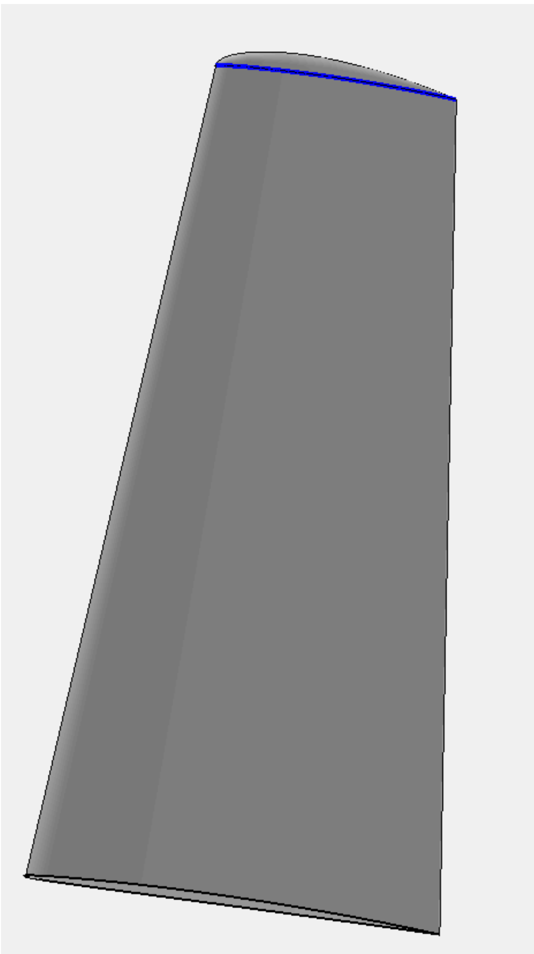
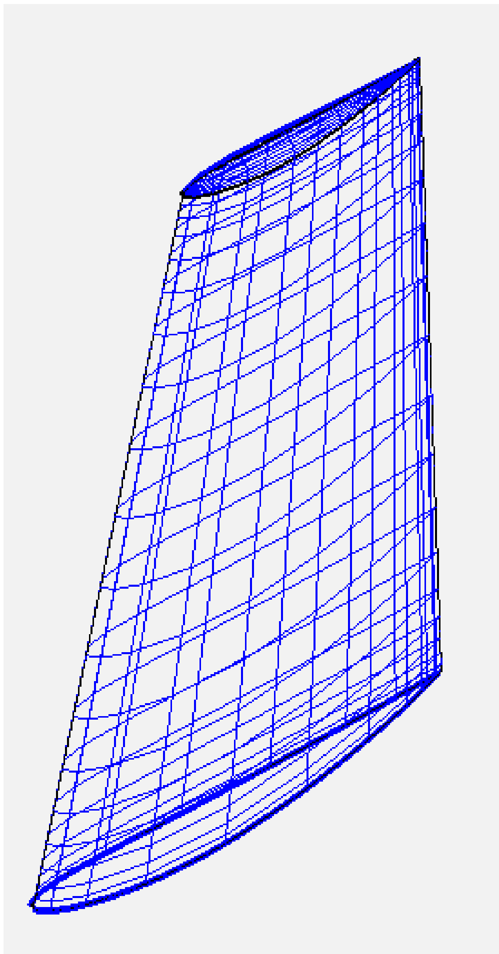
Parameters	Values
Span	4.918 m
Airfoil	NACA 4412
Area	3.615 m^2
Root chord length	1.207 m
Tip chord length	0.647 m
Aspect Ratio	6.691



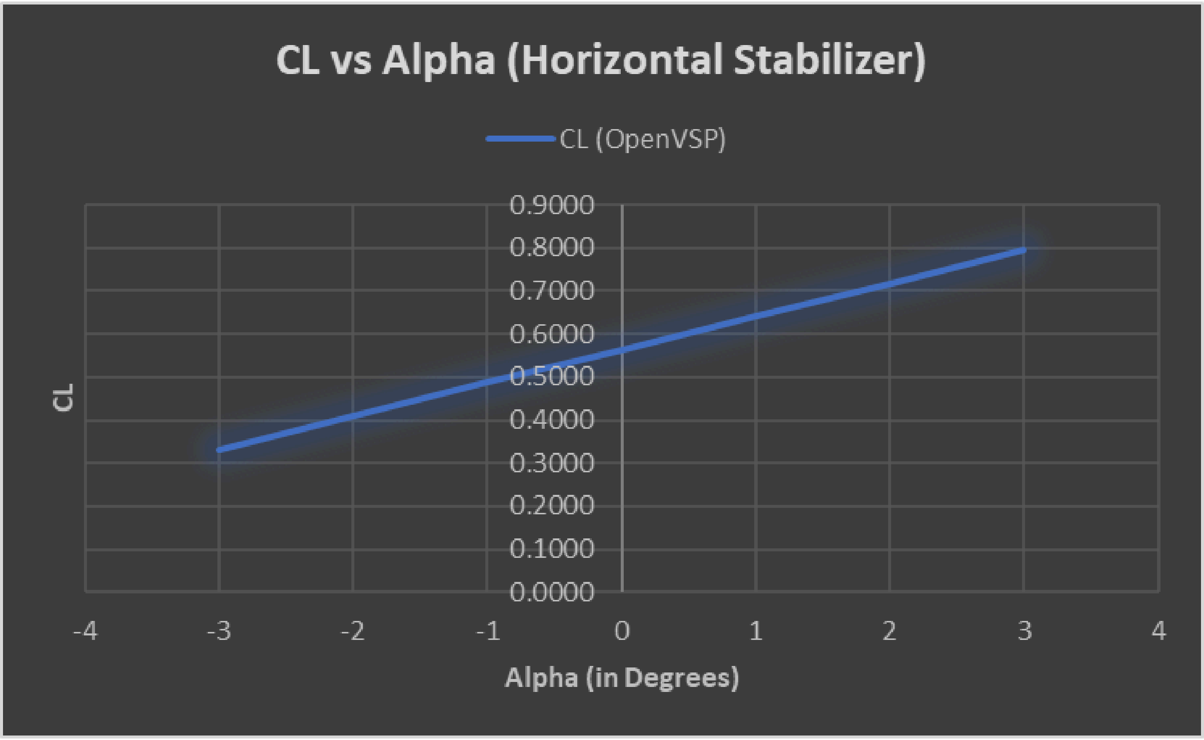


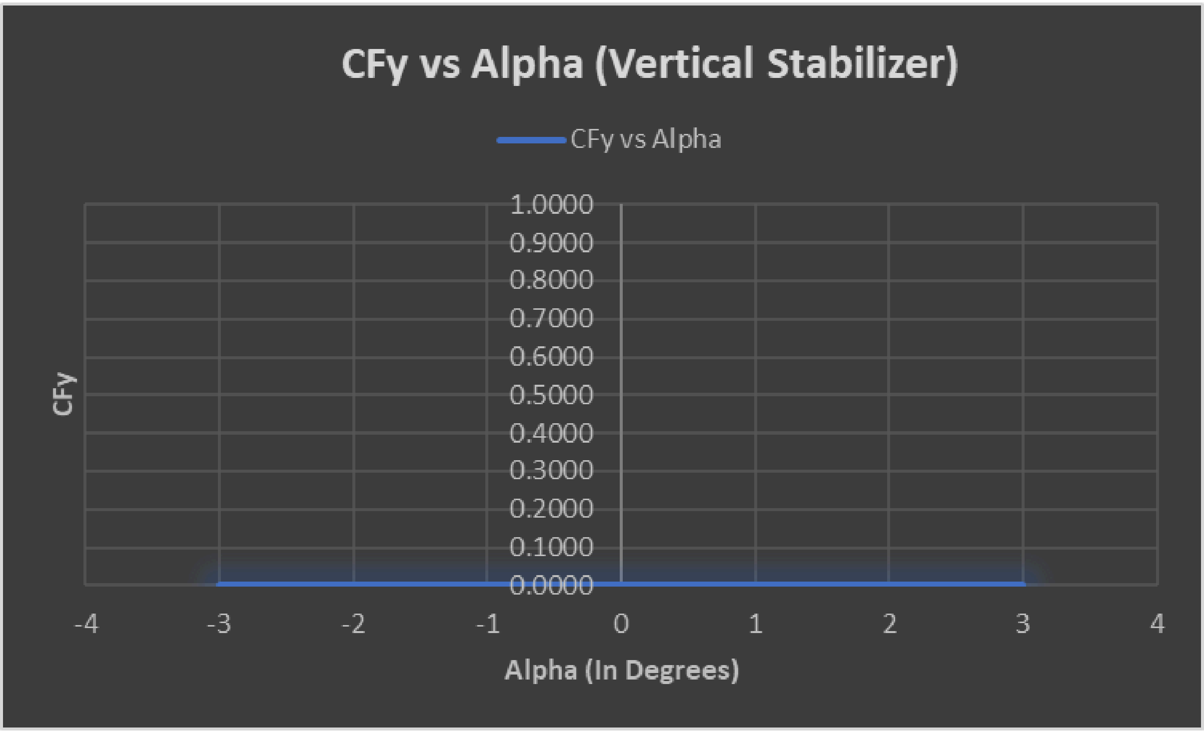
Vertical Stabiliser

Parameters	Values
Span	2.655 m
Airfoil	NACA 4412
Area	4.051 m^2
Root chord length	1.743 m
Tip chord length	0.841 m
Aspect Ratio	1.751

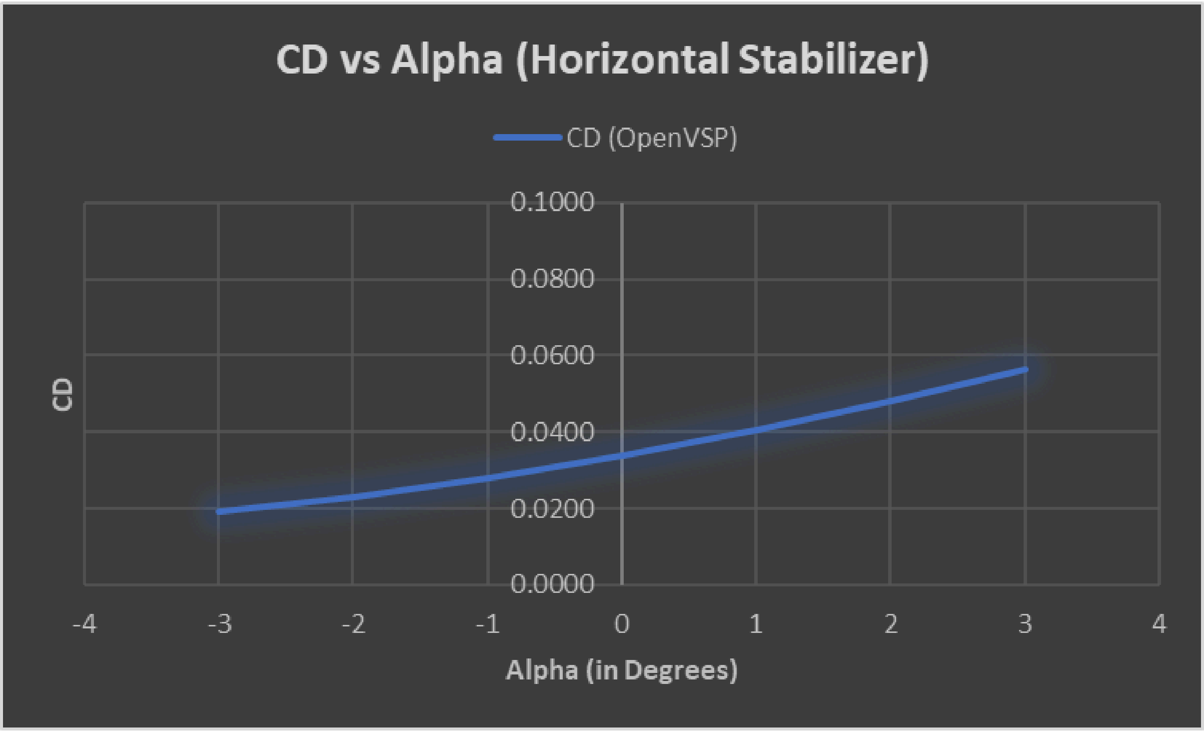
 

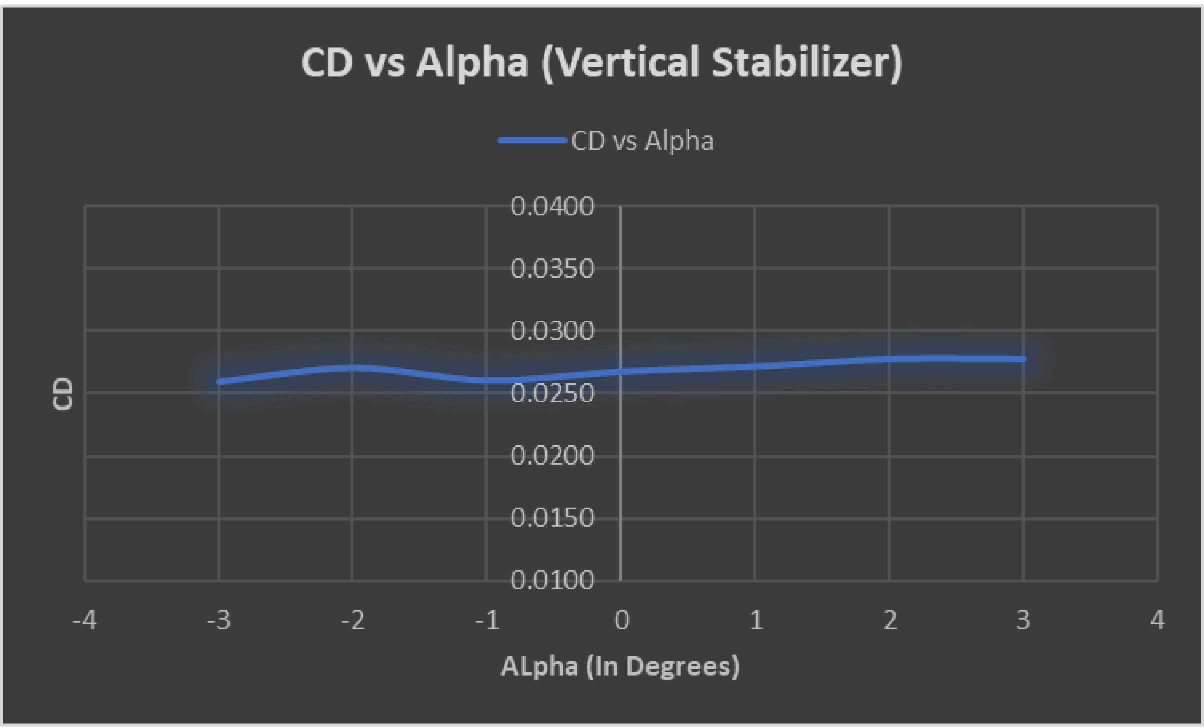
3.2 CL vs α curves (α = -3° to 3° using OpenVSP)





3.3 CD vs α curves (α = -3° to 3° using OpenVSP)





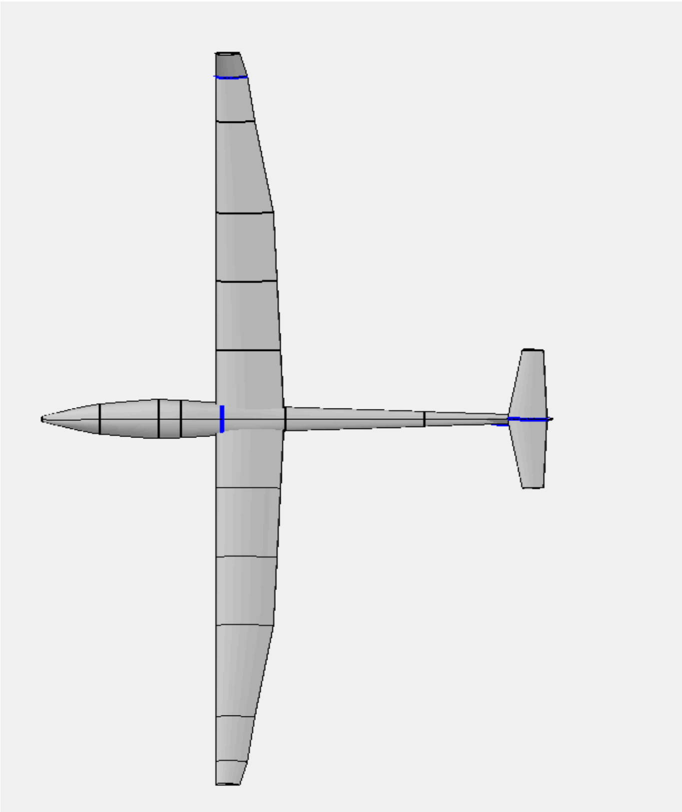
3.4 Comments on findings

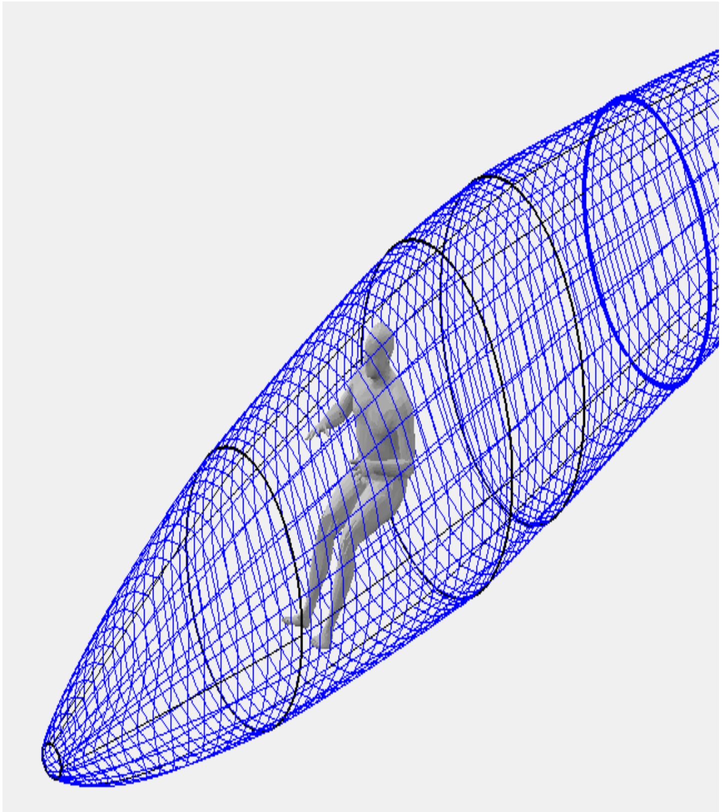
As we see we get a decent CL vs alpha curve for horizontal stabilizer but when we see the curve of CFy vs alpha curve for vertical stabilizer (As force is in only y direction for it so we consider CFy instead of CL values from OpenVSP) it values are almost zero on all angles as expected because obviously vertical stabilizer will not produce any significant lift at all.

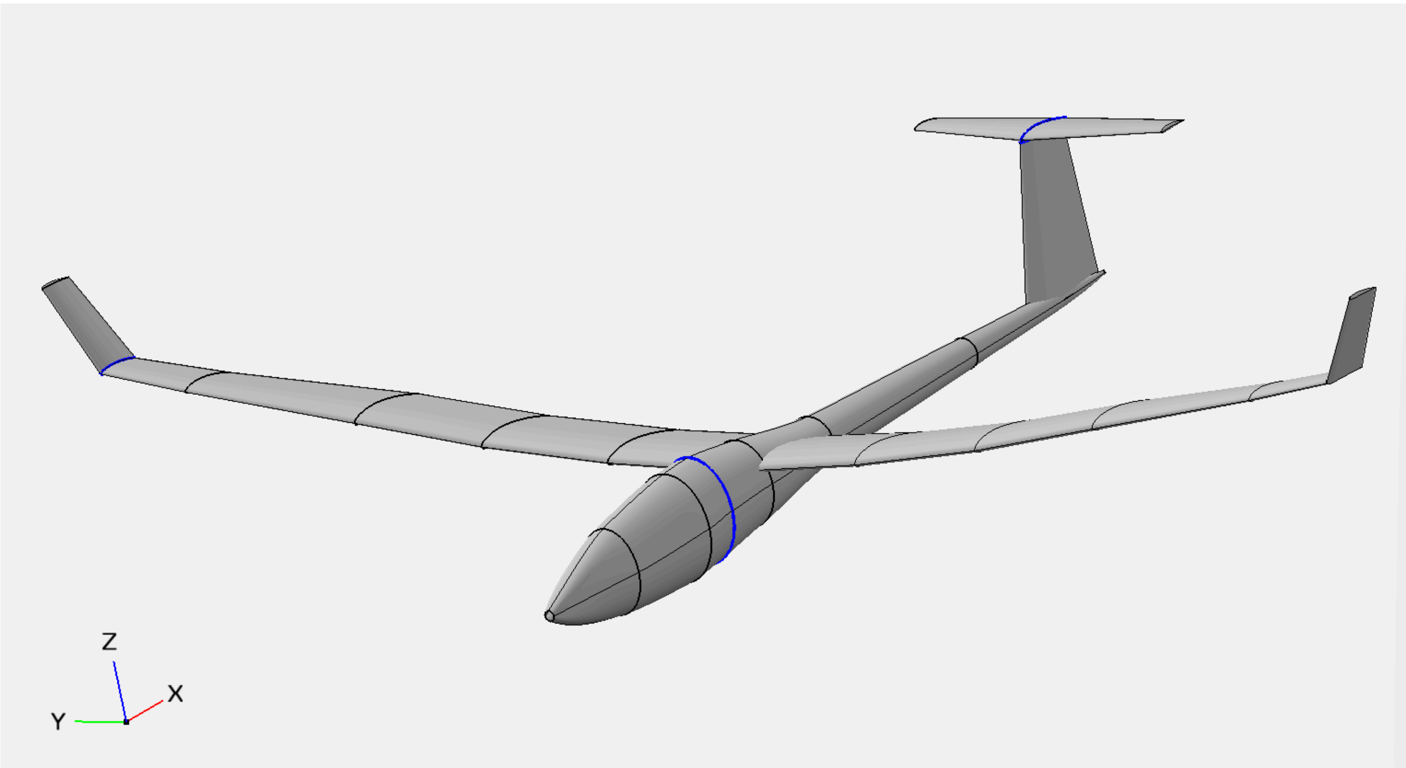
For CD vs Alpha curves, In horizontal stabilizer CD curve is of increasing slope but for the vertical stabilizer CD curve is almost of constant nature. Reason behind this is that only parasitic drag is present in the case of vertical stabilizer and there is no presence of induced drag.

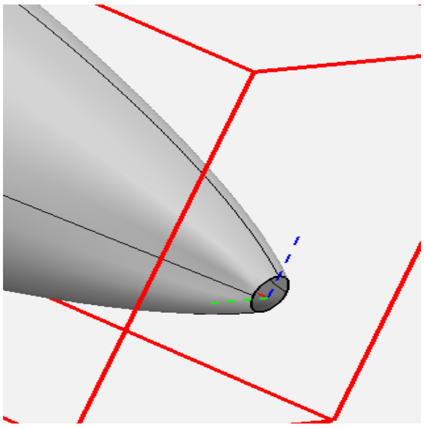
4. Overall Glider Design

4.1 CAD and Positions







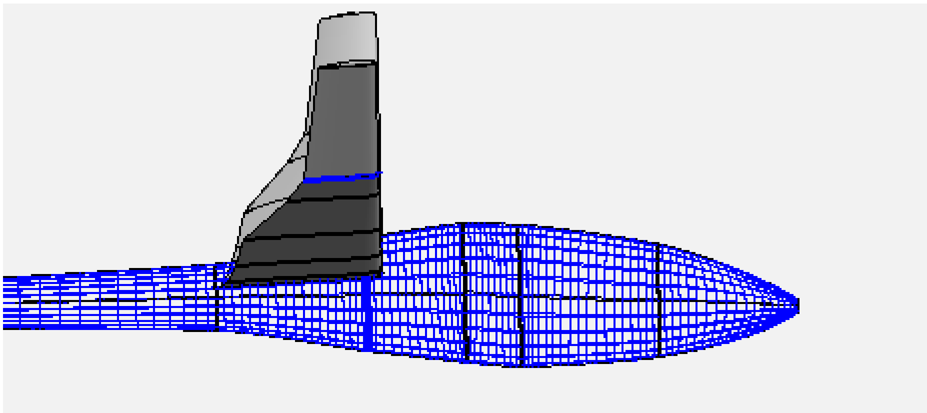


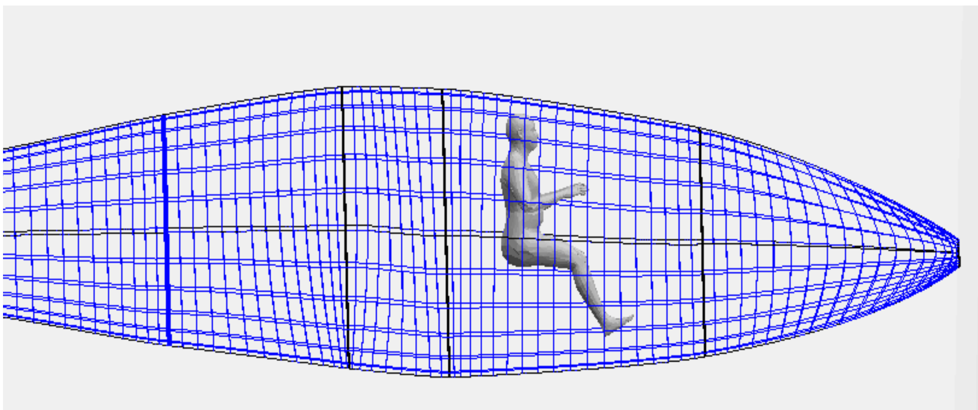
Origin is the tip of our Glider

(All Positions are with respect to origin)

Wing Position

X - 18 ft
Y - 0 ft
Z - 1.78 ft

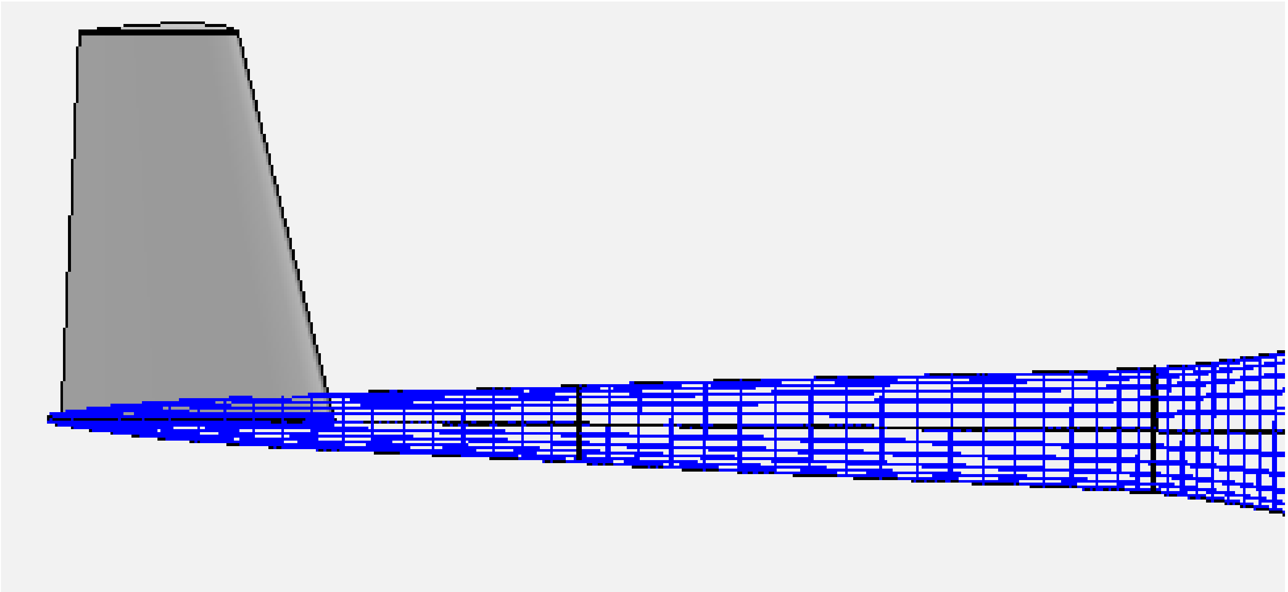




Human
X -10 ft
Y - 0 ft
Z - 0 ft

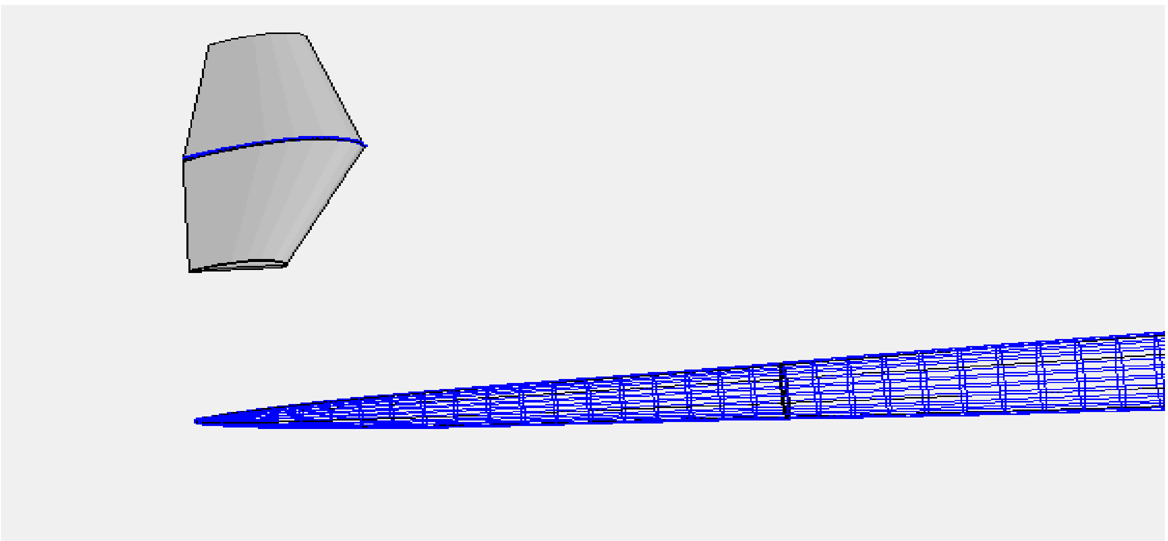
Vertical stabiliser

X - 44 ft (Starting)
Y - 0 ft
Z - 1.06 ft

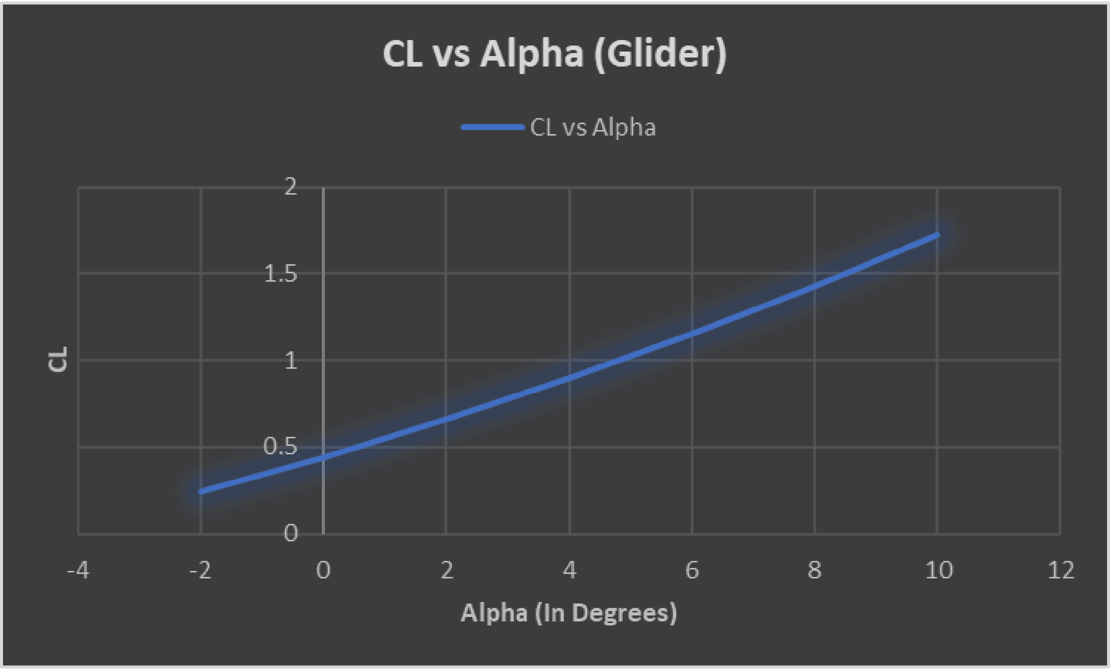


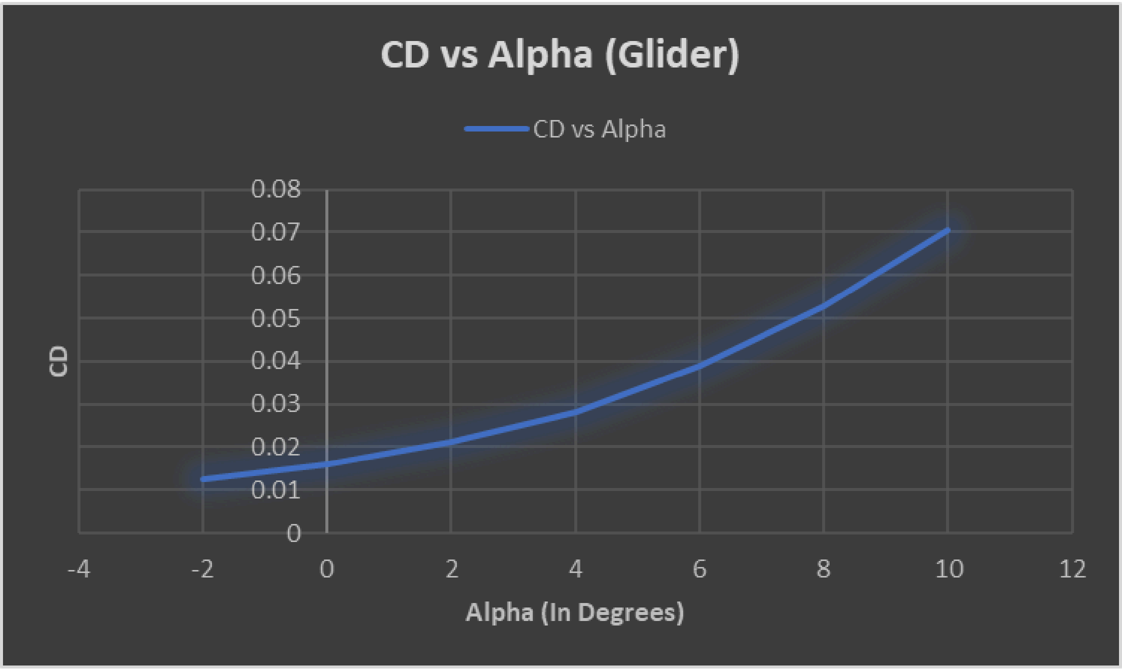
Horizontal Stabiliser

X - 50 ft
Y - 0 ft
Z - 8.76 ft



4.2 Performance of the entire glider (for α = -2° to 10°)





Observations-

1. As expected, we can see from the CL vs Alpha curve that the CL values for the glider are coming less than the CL values of the wing.

2. I noticed that almost 60-65 % of the total drag is coming from the wing only.

3. There is a slightly more divergence, particularly at higher angles of attack, around 8° and 10° for the CL and CD values of the glider as compared to the wing. This variance happens due to the influences from interactions between the fuselage and stabilisers.

4.3 Glider component weight estimates and total weight

Materials	Use	Components	Density (Kg/m3)	input	weight (in kg)
Glass Fibre	Use Surface area (3mm thickness)	Wings, Stabilizers, Fuselage	1100	92.16	330.128
Aluminium Spars	1/15 vol	Fuselage	2600	0.591	102.44
Aluminium Spars	1/20 vol	Wings, Stabilizers	2600	2.235	296.192
		Human			80
					808.766 Kg

Components	Surface area(m2)	Volume (m3)
Wing	44.01	1.953
Horizontal stabilizer	3.6	0.1964
Vertical Stabilizer	4.05	0.0857
Fuselage	40.45	0.591

Here we are not incorporating the weights of electrical components and other required components.
We are

After incorporating all the inputs like surface area and volumes of Wing, fuselage and stabilisers, We found that

Human Weight = 80 kg

Weight of Glider including Human is 808.766 Kg

W = 808.766 * 9.81 = 7934.85 N

5. Design Validation

5.1 Optimal speed, glide angle, range, descent rate for the designed glider

Velocity = 30 m/s

At AOA = 4° for glider,

Cl = 0.8974
Cd = 0.0296

$$q_o S_{ref} = 0.5 * 1.225 * 30 * 30 * S_{ref} = 551.25 * 16.18 = 8919.22$$

$$L = q_o S_{ref} * Cl = 8919.22 * 0.8974 = 8004.108 \text{ N}$$

$$D = q_o S_{ref} * Cd = 8919.22 * 0.0296 = 264.009 \text{ N}$$

$$L/D = 8004.108 / 264.009 = 30.31$$

So, here

L/D > 30

So, the Gliding Condition is satisfied.

Now for weight condition,

$$W = \sqrt{L * L + D * D}$$
$$W = \sqrt{8004.108 * 8004.108 + 264.009 * 264.009} = 7942.64 \text{ N}$$

So, **7942.64 N ≈ 7934.85 N**

Hence Weight condition is also verified.

Angle of Attack	**4°**
Velocity	**30 m/s**
Glide Ratio	**30.63 : 1**
Glide Angle	**1.88°**
Rate of Descent	**0.984 m/s**
Range (Release Height = 1km)	**30.63 km**
Weight	**809 kg**

5.2 Comments and Scope of Improvements

1. The current glider design surpasses the required range, but there's potential for further improvement.

2. The fuselage length exceeds the necessary specifications and could benefit from further optimization.

3. Additionally, optimising the moment balance would enhance the positioning of the stabilisers. The fuselage currently generates some downforce, indicating room for optimization.

4. Furthermore, the design of the stabilisers needs refinement as it has only been roughly outlined thus far.

6. Acknowledgement

1. Ghoshank Nanhe (22b0073)
2. Nikhil Jha (22b0002)
3. Devesh Mittal (22b0070)
4. Chaitanya Keshri (22b2472)

7. References

1. https://students.iitk.ac.in/aeromodelling/downloads/glider.pdf
2. https://vspu.larc.nasa.gov/
3. https://youtu.be/ilOXnWJNsDc?si=iBbLOY_3RCfAAJ0V
4. https://youtu.be/iiEMm7P2skI?si=hMgz_tgaC7wit3HW
5. https://youtu.be/SO3SWw92xj0?si=42Om2WajoHnSbfqW
6. AE244 Low speed Aerodynamics Notes