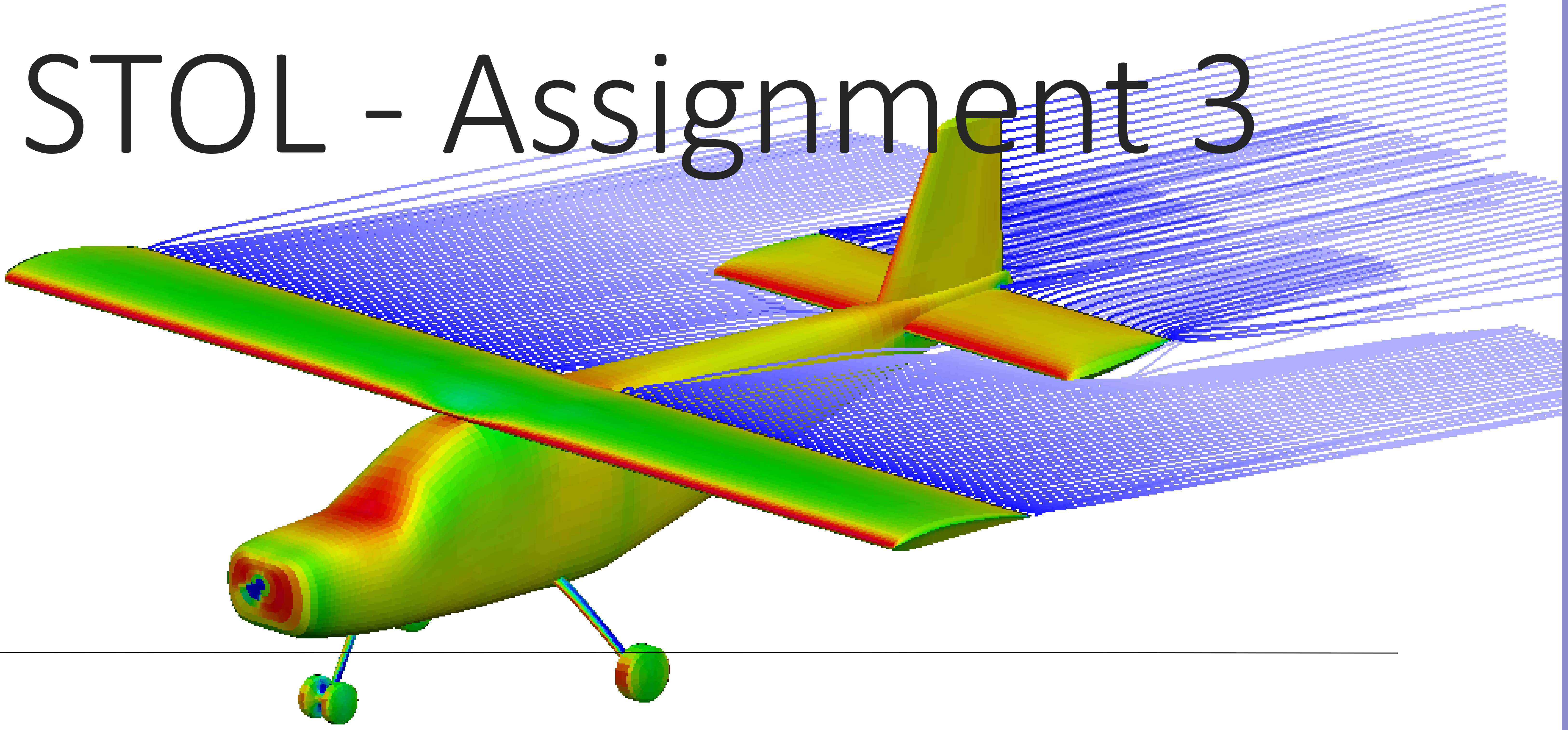
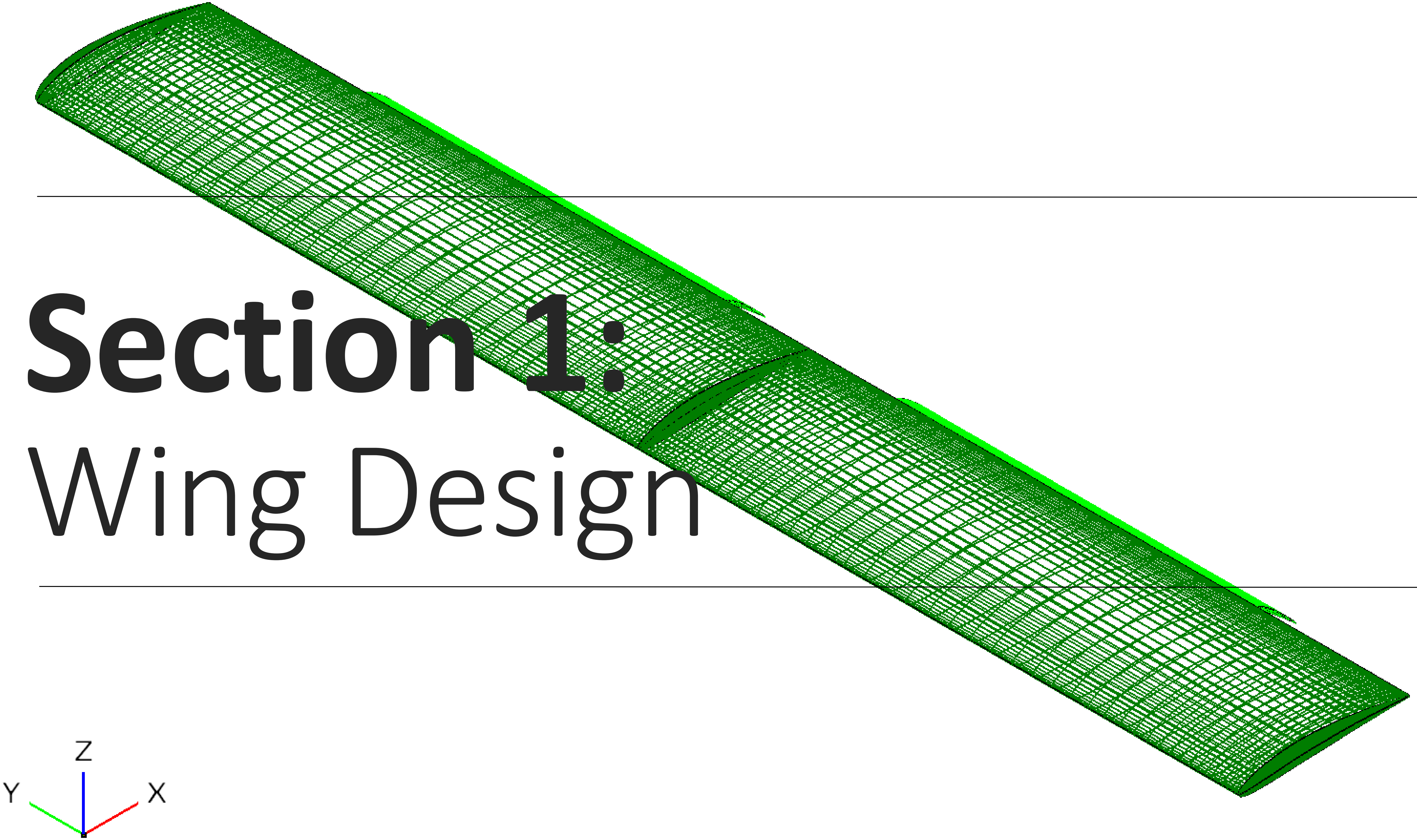
STOL - Assignment 3



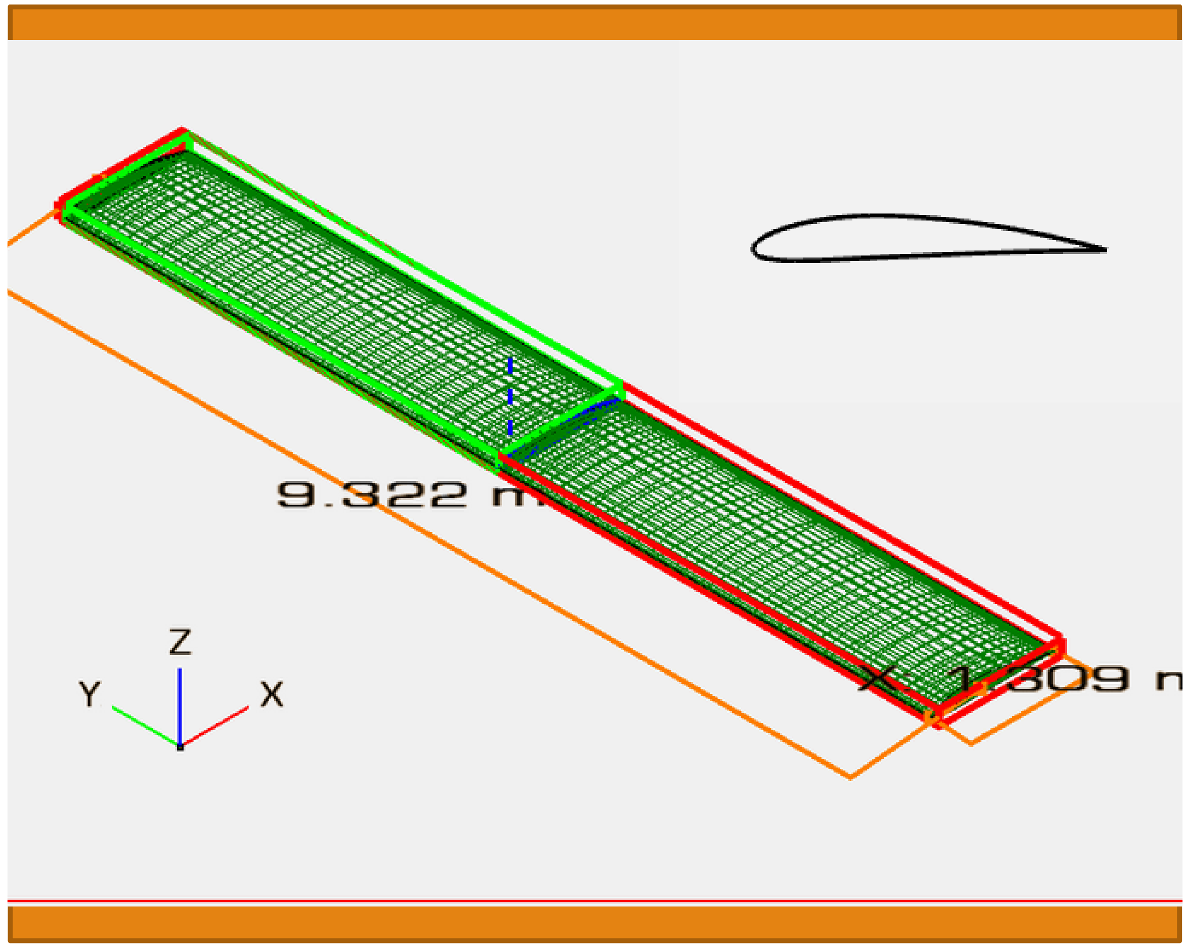
Name: OM NUNASE
Roll Number: 23B0003

Section 1:
Wing Design

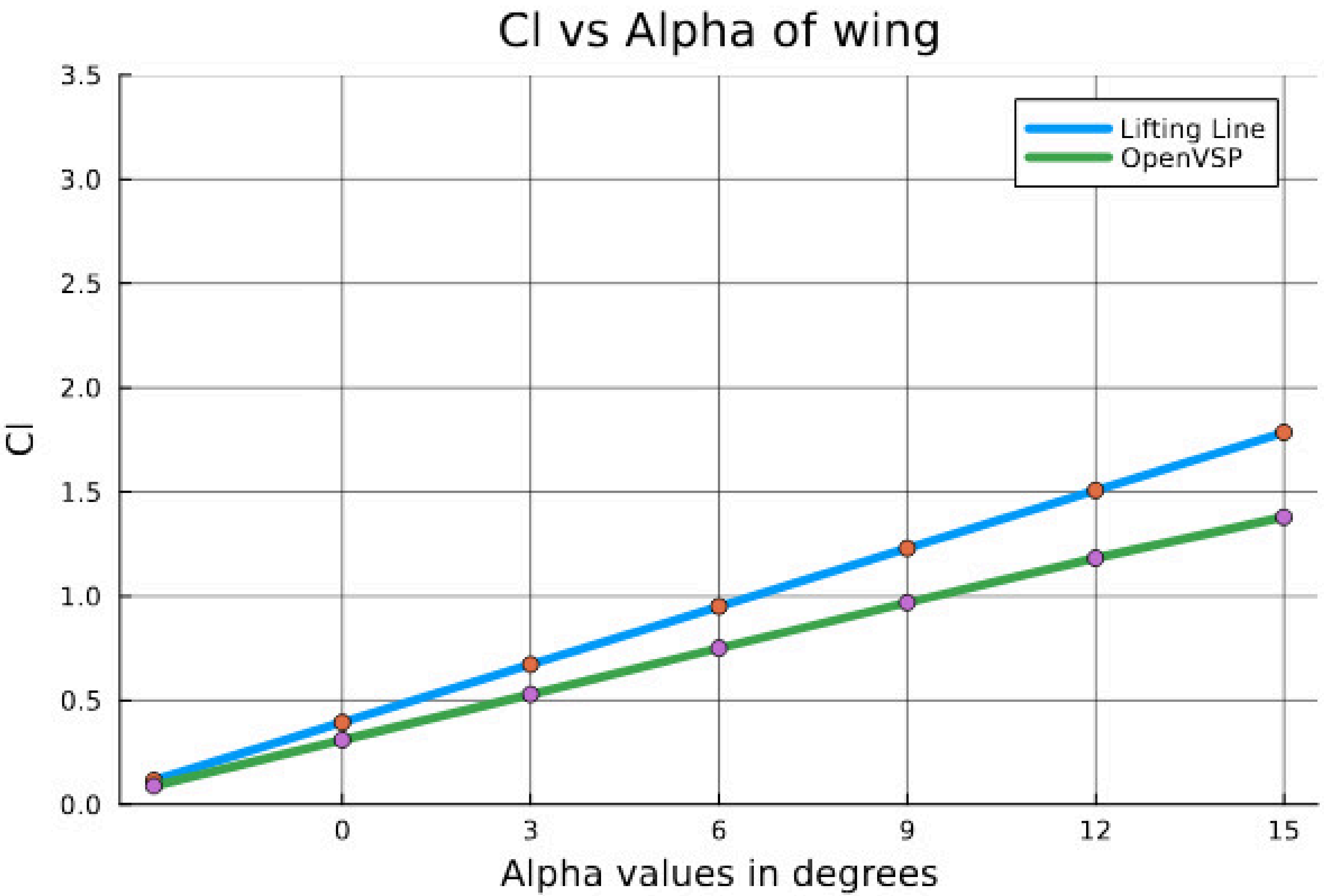


1.1:
Wing Design

Design Parameter	Value/ Description	Rationale
Airfoil (s)	NACA 4412	The given airfoil provided adequate CL at zero degrees and also when equipped with flaps, gave CL = 2.5 at 12 deg. AoA, sufficient to takeoff. Also, the standard data was available for this airfoil to compare the OpenVSP results.
Chord Distribution	uniform chord throughout chord length = 1.309m	rectangular wings are easier and cheaper to manufacture, given the face that we might be already constrained by the use of carbon fibre
Span	9.322 m	derived using the code
Pitch angle distribution	1.9 deg. (constant throughout)	In order to cancel out the net torque and net force at cruise.
Sweep	0	NO sweep needed in order to maximize the lift
Dihedral	0	maximize lift
Fowler Flap	max deflection: 40 degree (wrt absolute)	I tried various angles, this one gave me the required CL for takeoff at the appropriate angle of around 14 degree.
Hoerner wingtip	At the end, there is a sharp edge, known as Hoerner wingtip.	It provides adequate increase in lift and has a very low mass penalty.

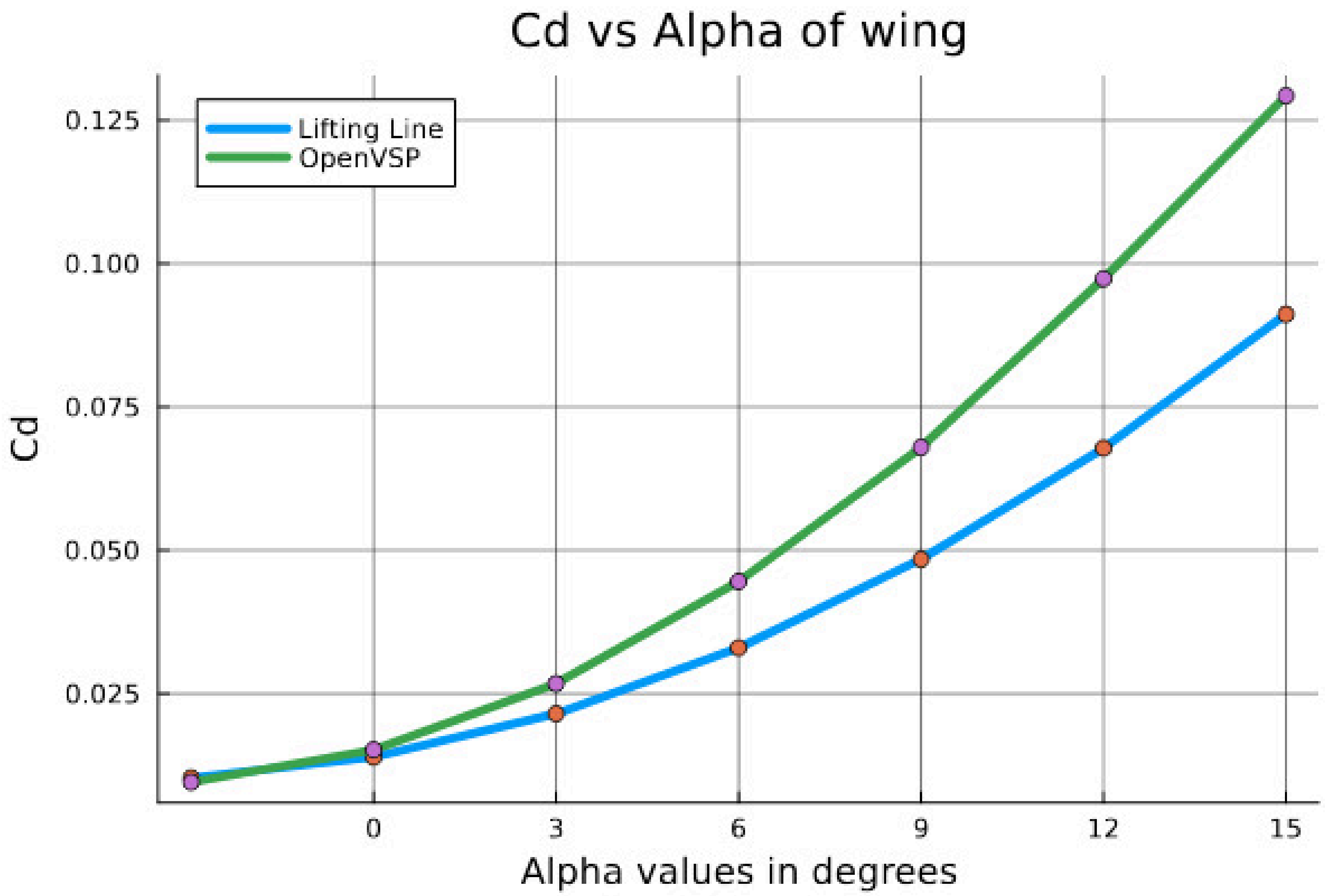


1.2: CL vs α



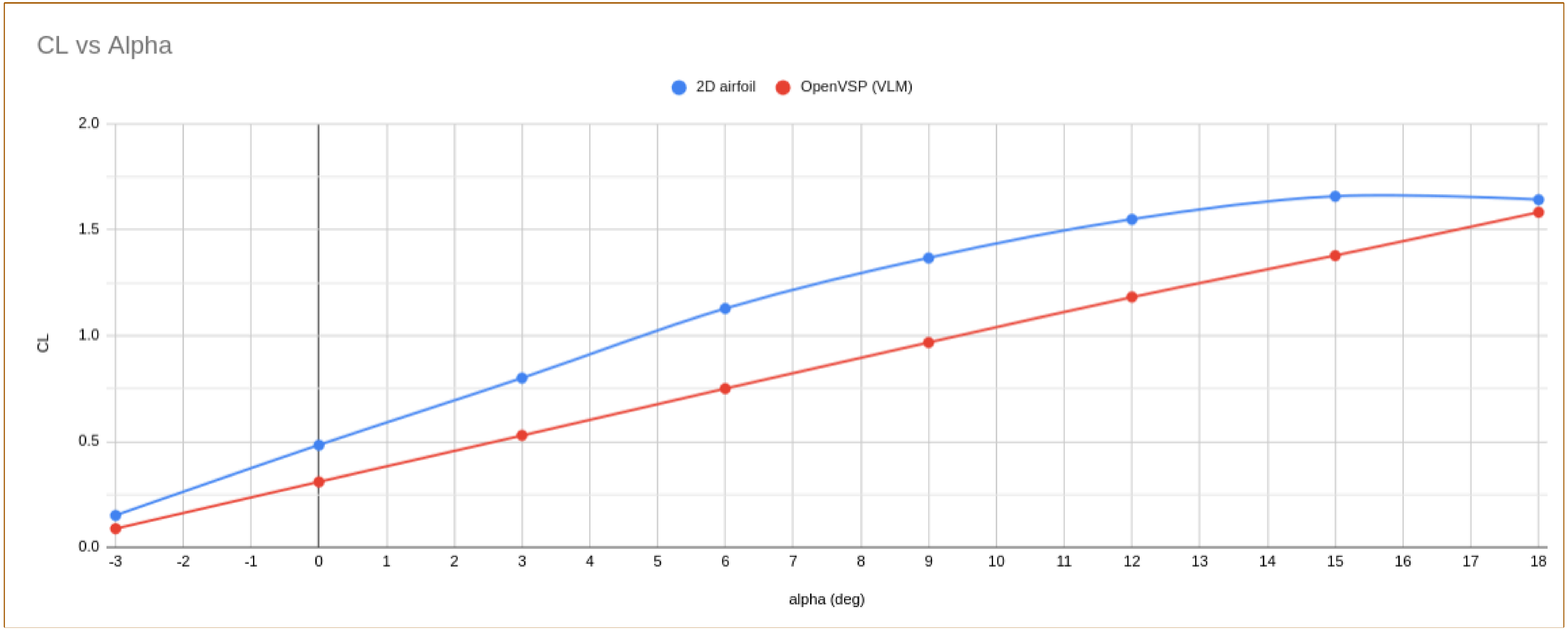
Cl vs Alpha of wing
Lifting Line
OpenVSP
3.5
3.0
2.5
2.0
1.5
1.0
0.5
0.0
Cl
0
3
6
9
12
15
Alpha values in degrees

1.3: CD vs α



Cd vs Alpha of wing
Lifting Line
OpenVSP
0.125
0.100
0.075
0.050
0.025
Cd
0
3
6
9
12
15
Alpha values in degrees

1.4:
CL vs α (Wing vs Airfoil) {no flaps}



1.5: Main Observations and Interpretations

CL vs alpha:

- CL increases linearly with
- We can see that the values are approximately equal for small angles. but as angle of attack increases, we see a deviation in the graphs. This is because lifting line theory works well for small angles only, while VLM considers full 3d effects like sweep, twist, taper, etc... and an accurate estimation of the downwash velocity.

CD vs alpha:

- CD increases non-linearly with angle of attack
- OpenVSP predcits higher drag values as compared to LLT at larger alpha because Openvsp condisers induced drag due to 3d effects, whereas LLT assumes idealised conditions with lower drag.

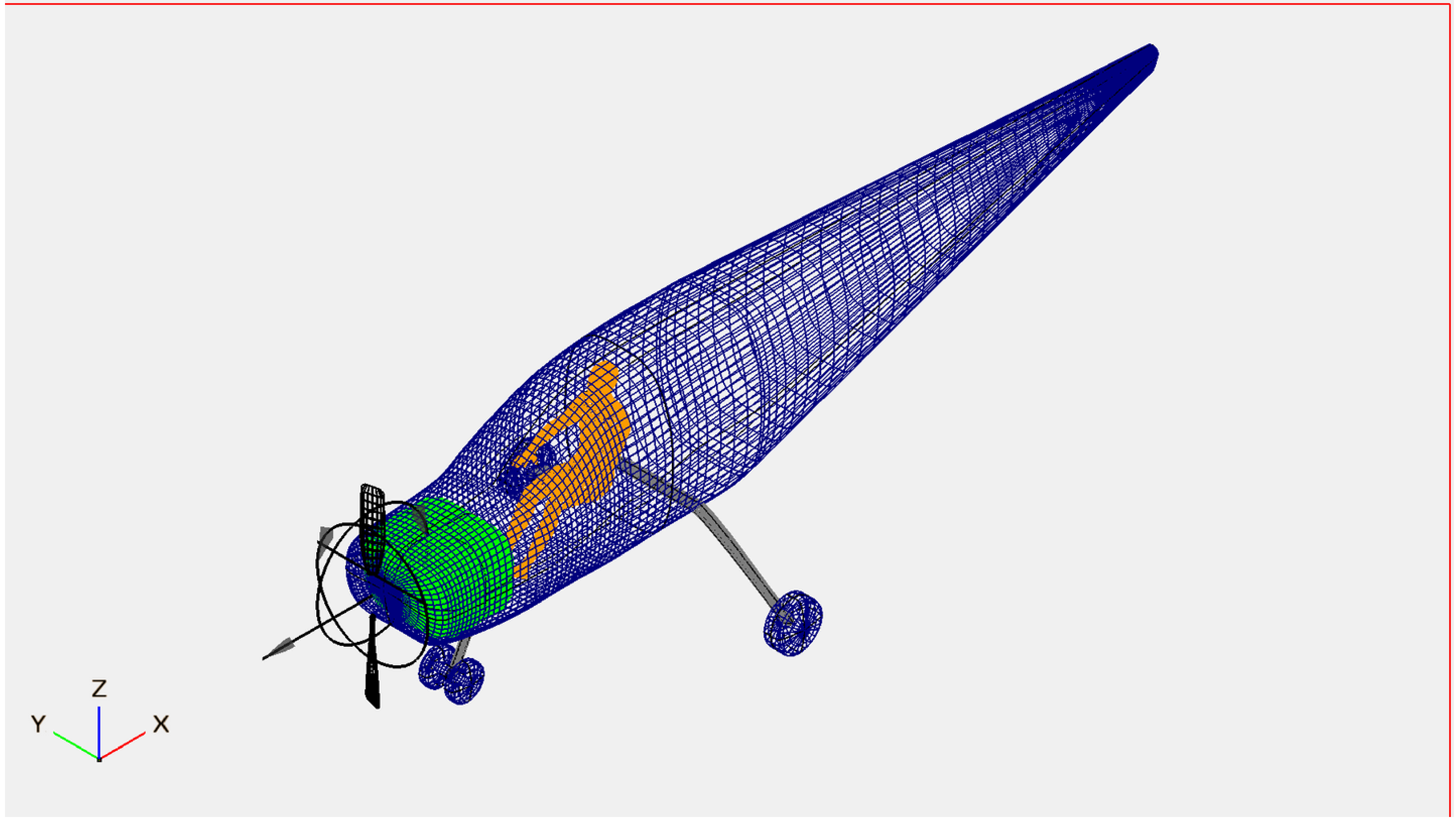
CL vs alpha (Wing vs 2D airfoil, no flaps)

- The Cl values are consistently lower than those of 2D airfoil
- This is due to 3D effects such as tip vortices and downwash that reduce the effective angle of attack over the wing.
- 2D airfoil results represent idealised conditions withouth spanwise flow and induced drag
- OpenVSP accounts for these factors, which leads to reduced CL values for the wing.

Section 2:

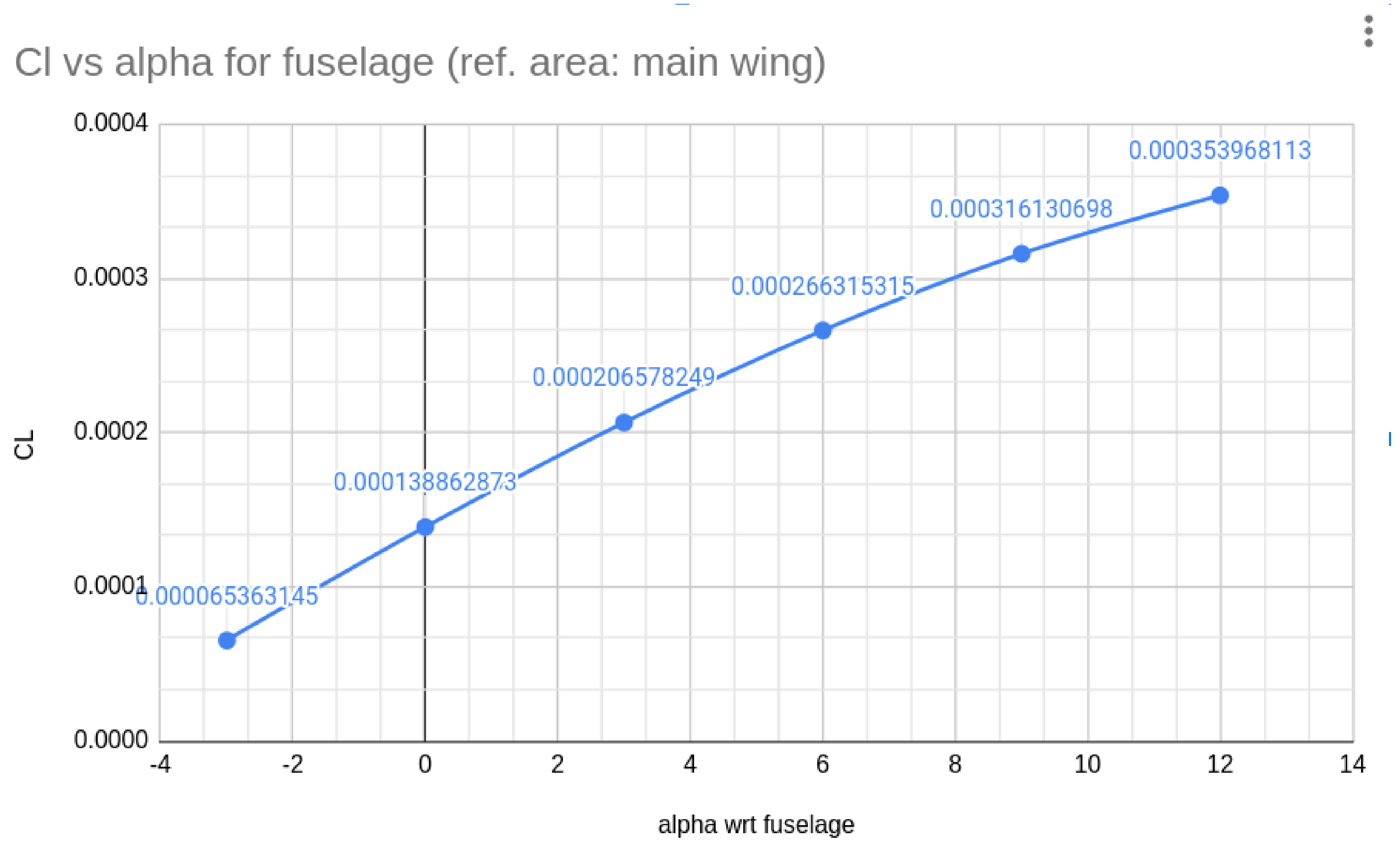
Fuselage Design

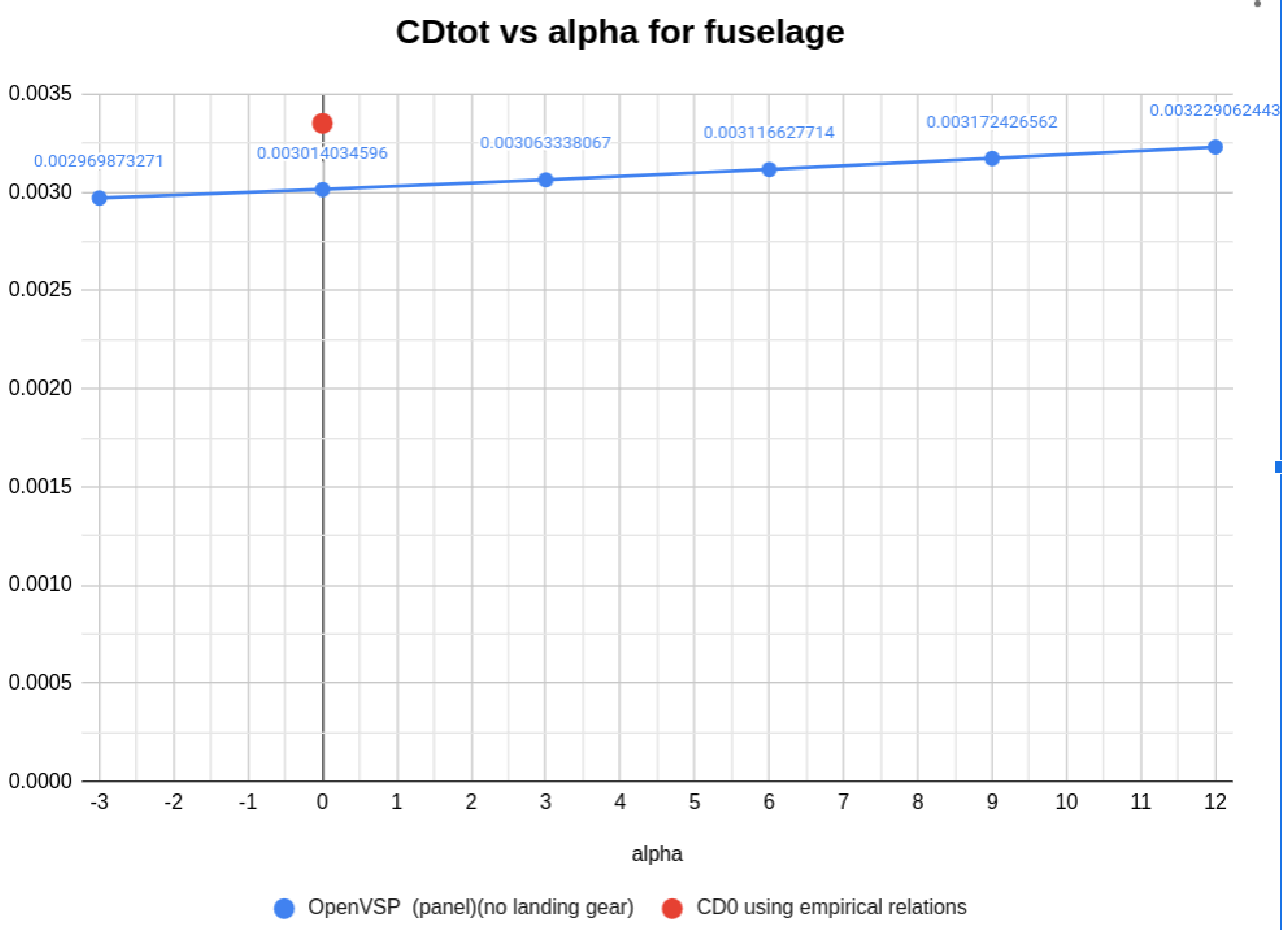
2.1: Fuselage Design



Design Parameter / Consideration	Value/ Description	Rationale
Length	7.0 m	initially: 0.8 x wingspan is almost 7 m. [code] When balancing forces and torques, it was found that 7 m is good enough length and so I did not changed.
Height	1.29 m (cabin height)	Pilot and seats fit inside this
Width	1.0 m (cabin width)	1 m is quite comfortable for me, too high and drag increases, too low and comfort decreases. Plus, engine needs 0.8 m width and so 1 m is kept as good margin.
other dimensions can be seen in the openvsp file		I have kept in mind the pilot's (my) comfort and space saving. As well as keeping the engine dimensions in mind, the front has been designed. Yes, the green part in the front is the engine. Nose landing gear is attached to the place where engine is attached, because its a structurally strong member.

2.2, 2.3:
CL & CD vs α (used panel method)





2.4: Main Observations and Interpretations

The reference area for aerodynamic calculations is taken as the main wing area, which is standard practice to normalize coefficients like CL and CD.

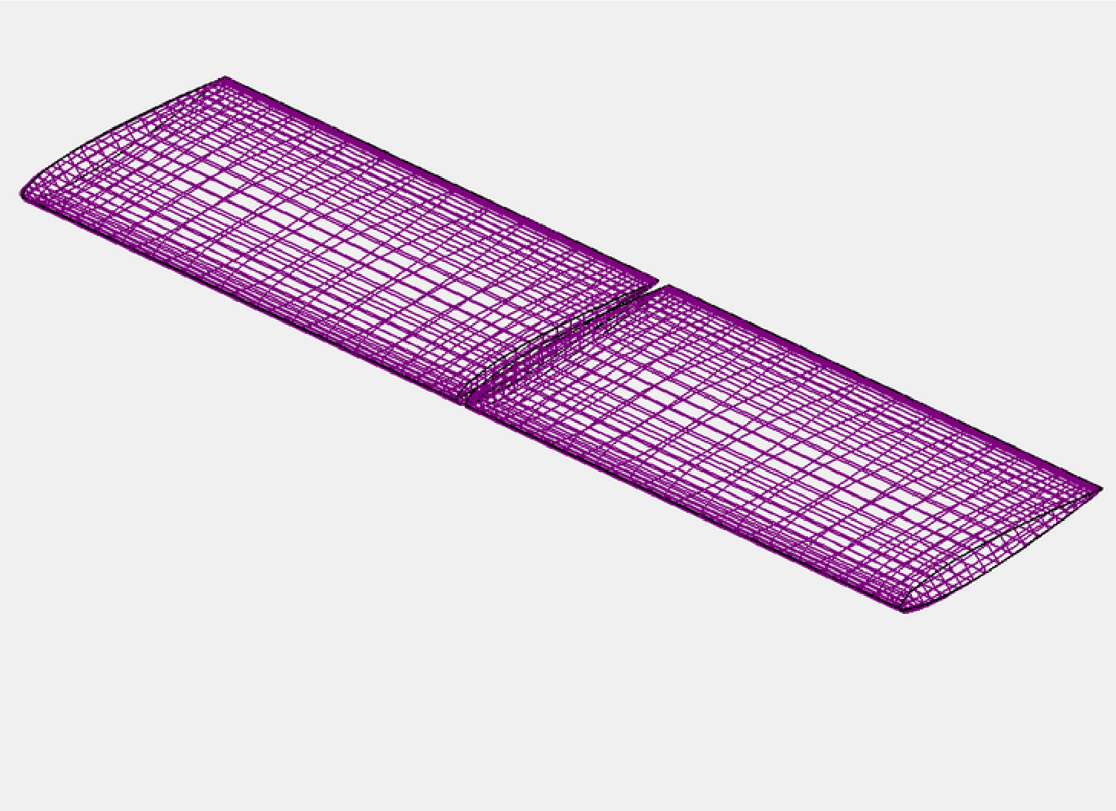
1. Lift Coefficient (CL):
   - The CL values are significantly lower compared to any wing. This is expected because the fuselage primarily contributes to drag and minimal lift.
   - As the angle of attack (AoA) increases, CL increases slightly but remains very small, indicating that the fuselage does not generate substantial lift.
2. Drag Coefficient (CD):
   - The total drag coefficient (CDtot) increases steadily with AoA. This is consistent with the fact that higher AoA increases pressure drag on the fuselage.
   - Empirical relations were used to validate CD values obtained via the Panel Method, show good agreement.
3. Panel Method Performance:
   - The Panel Method works well for simulating fuselage aerodynamics. I encountered issues initially with it but then i changed the position of landing gears a bit so it can overlap better and make a good mesh. It worked!
4. Interpretation:
   - The fuselage contributes minimally to lift but significantly to drag. Thus it is important to streamline its design for greater STOL performance.
   - Combining fuselage results with wing-flap simulations (using VLM) will provide a complete aerodynamic picture of the aircraft.

Section 3:
Stabilizer Design

3.1: Stabilizer Design

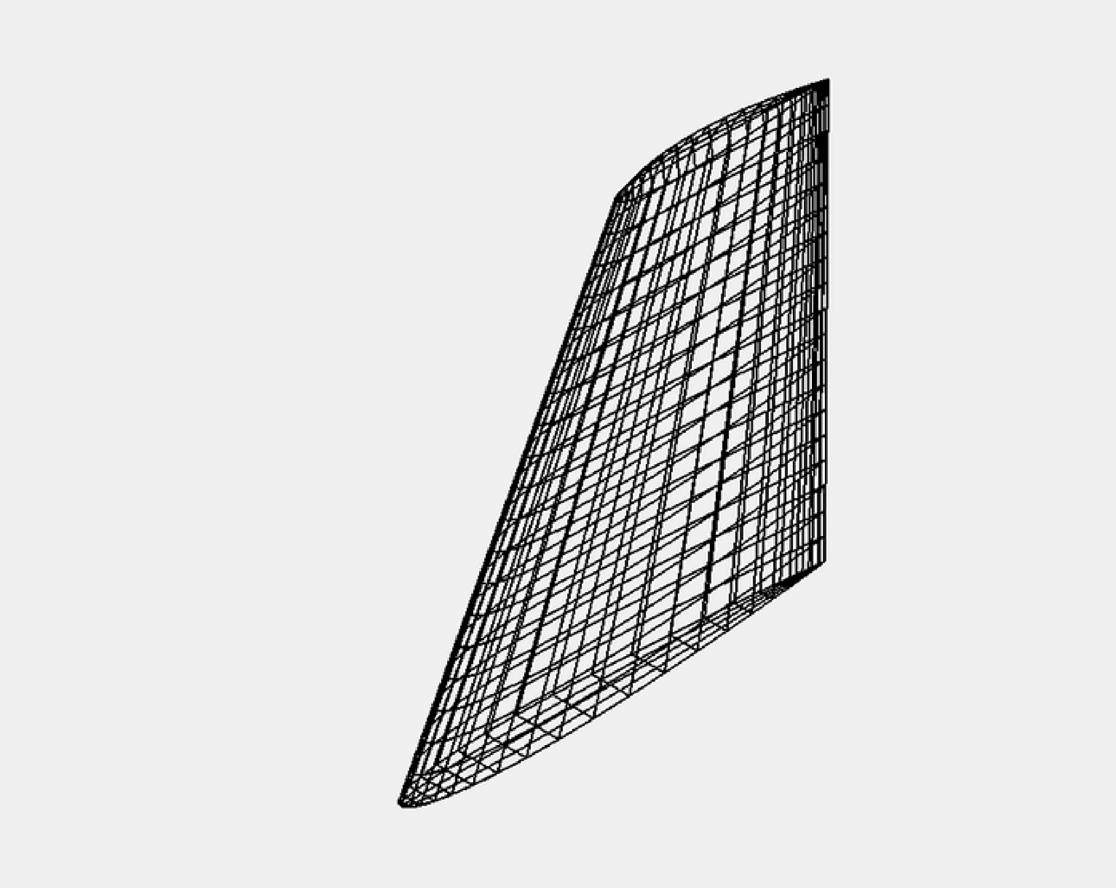
Horizontal Stabilizer

Design Parameter	Value/Description	Rationale
Airfoil	NACA 0012	A symmetric airfoil is best suited for the tail because of ease of control laws, as well as design. it has got predictable aerodynamics.
Chord Distribution	0.987m (uniform)	this simplies both the design as well as analysis
Span	3.924m	using empirical relations given in code.
Sweep	0	Sweep isn’t required for the speed regime we are operating in
Mounting angle	-5°	I balanced out the torques at cruise and this was the angle which the elevator should be placed in order to give the necessary lift.

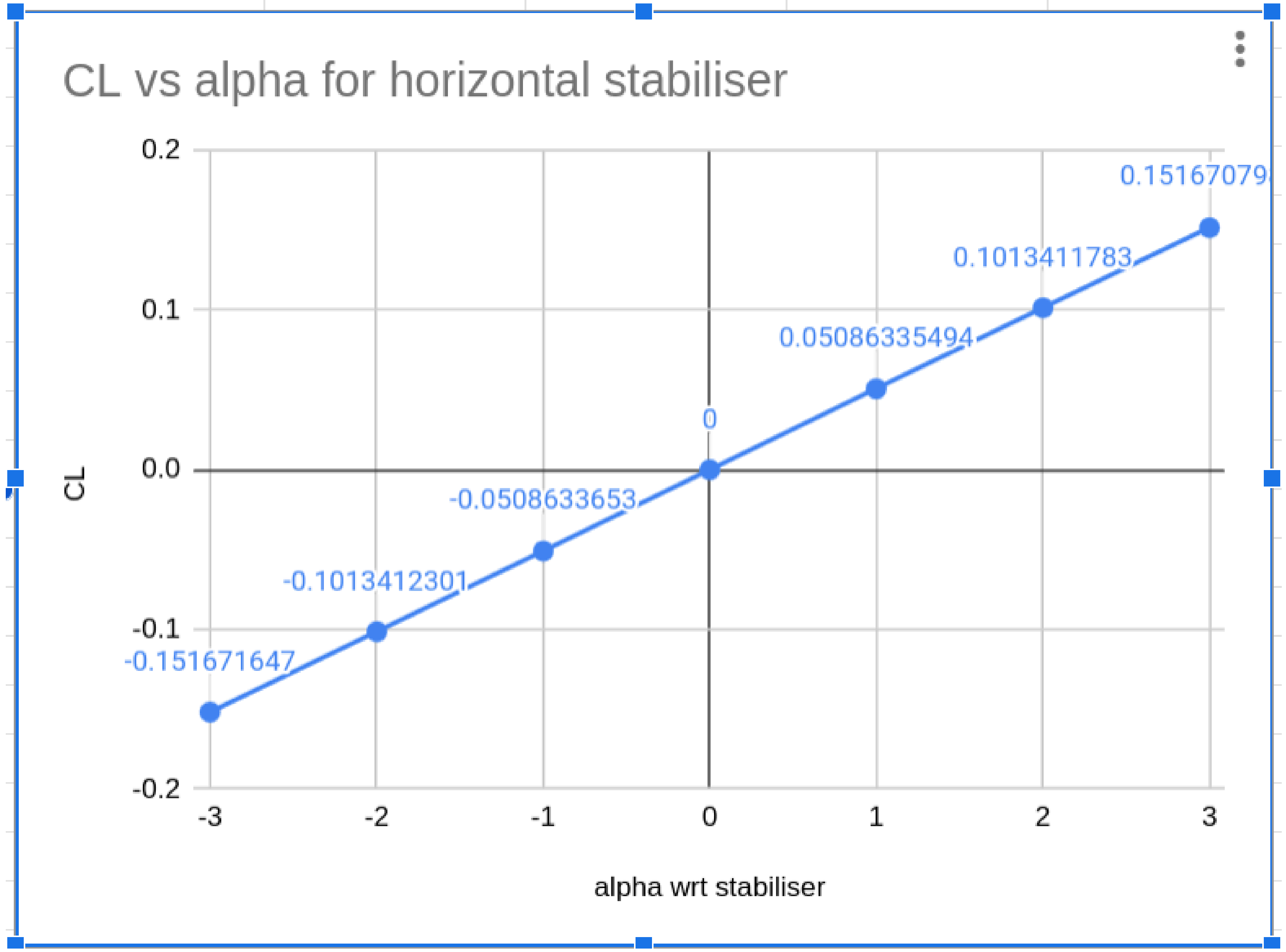


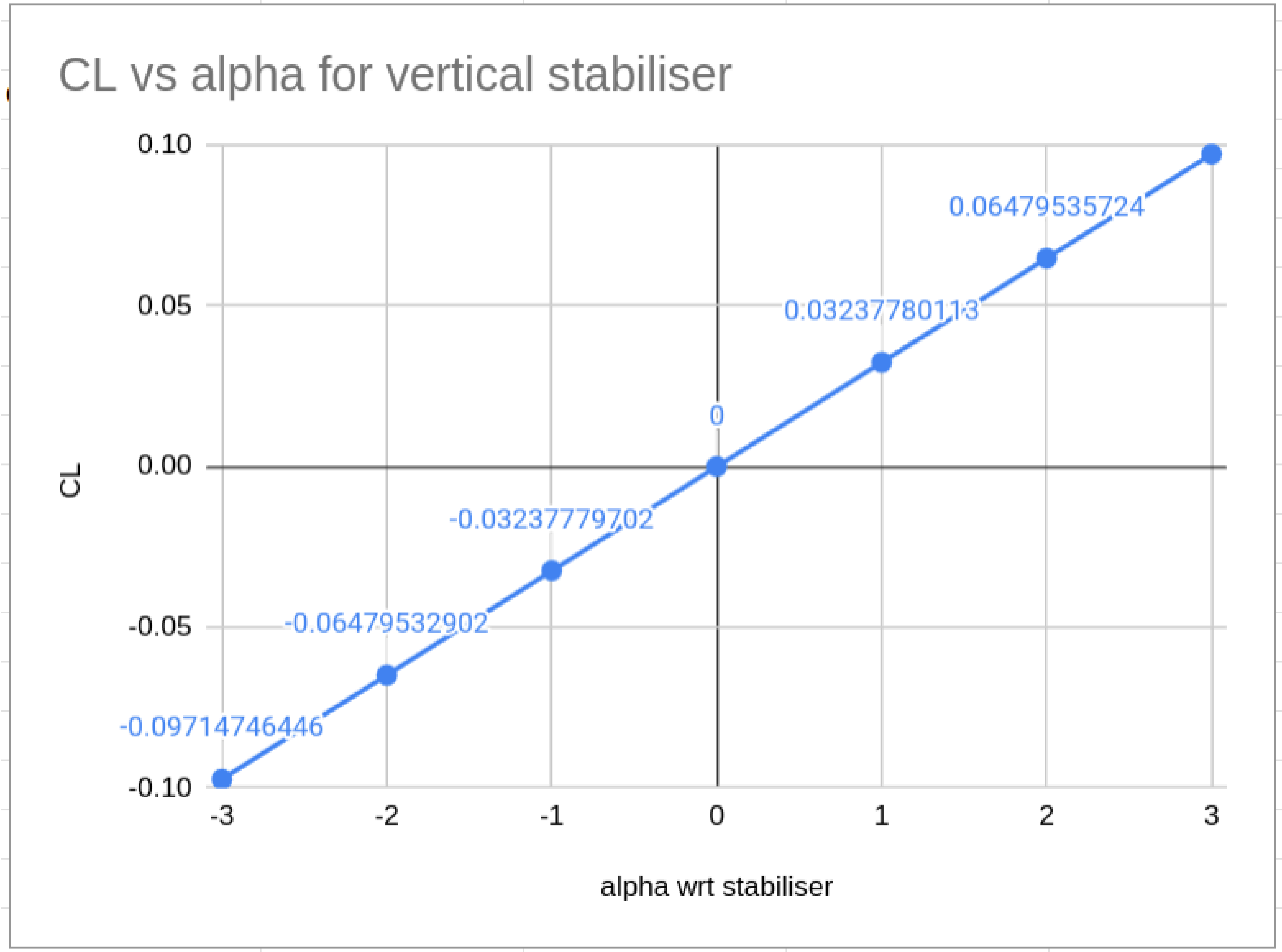
Vertical Stabilizer

Design Parameter	Value/Description	Rationale
Airfoil	NACA 0010	A thinner airfoil to minimize drag
Chord Distribution	root chord = 0.6m tip chord = 1.2m	to provide the necessary area
Span	1.18 m	To provide the necessary area
Sweep	20°	A swept vertical tail helps delay airflow separation at higher angles of attack during cross wind landings and takeoff. It looks cool also.



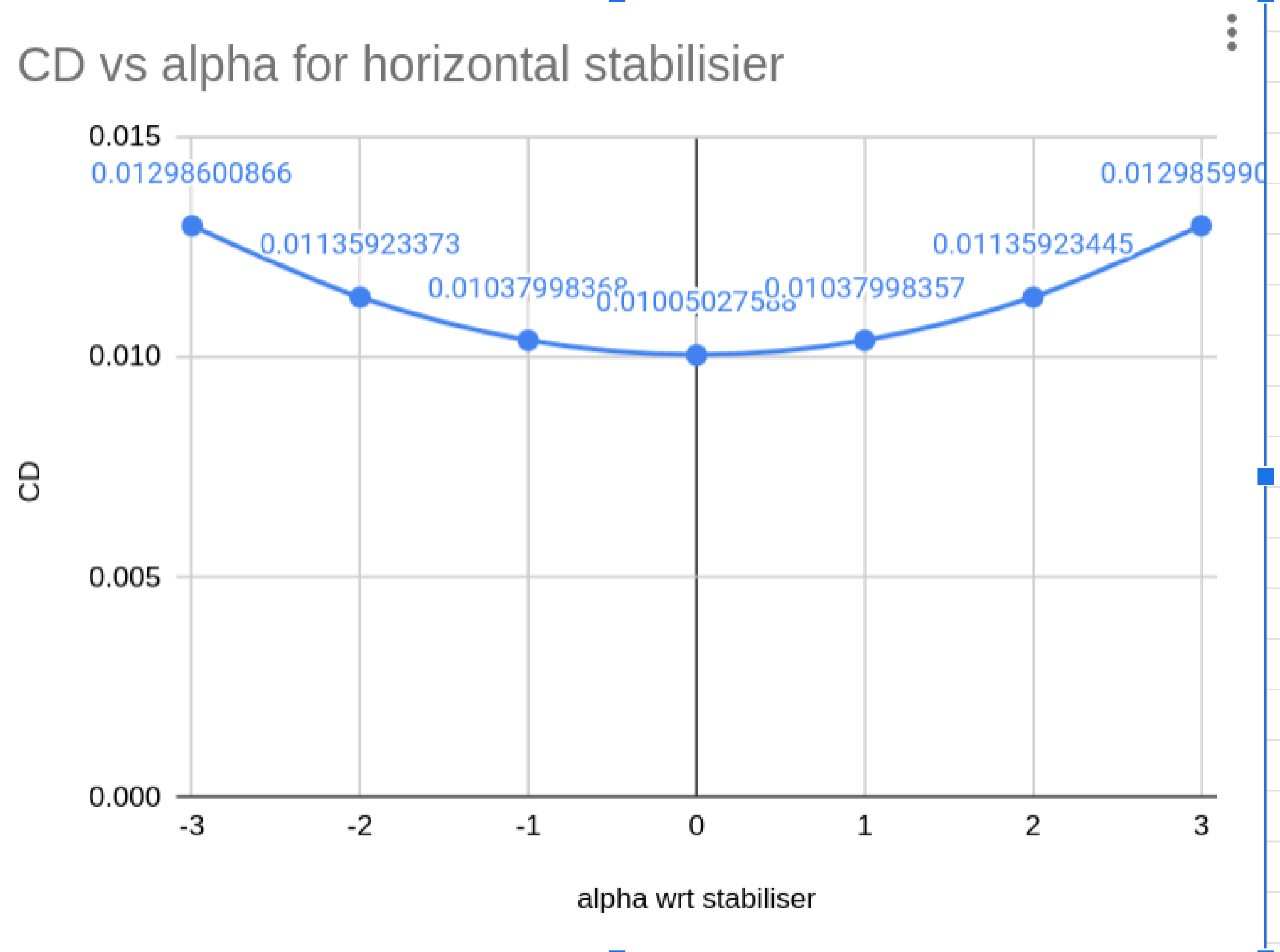
3.2:
CL vs α


CL vs alpha for horizontal stabiliser
0.2
0.1
0.0
-0.1
-0.2
CL
0.15167079
0.1013411783
0.05086335494
0
-0.0508633653
-0.1013412301
-0.151671647
-3
-2
-1
0
1
2
3
alpha wrt stabiliser

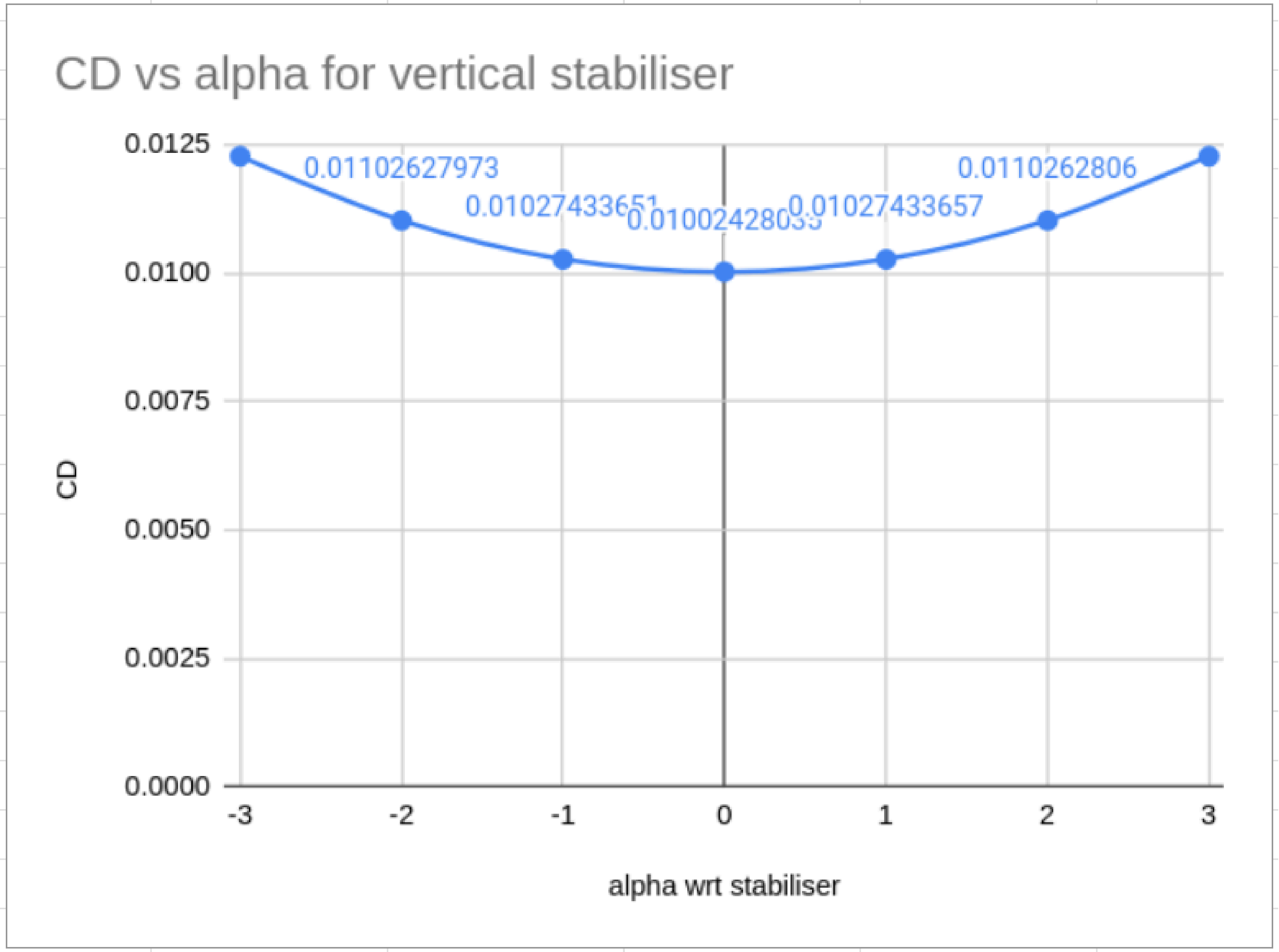

CL vs alpha for vertical stabiliser
0.10
0.05
0.00
-0.05
-0.10
CL
0.06479535724
0.03237780113
0
-0.03237779702
-0.06479532902
-0.09714746446
-3
-2
-1
0
1
2
3
alpha wrt stabiliser

3.3:

CD vs α



CD vs alpha for horizontal stabilisier
0.015
0.010
0.005
0.000
CD
0.01298600866
0.01135923373
0.01037998357
0.01135923445
0.012985990
-3
-2
-1
0
1
2
3
alpha wrt stabiliser



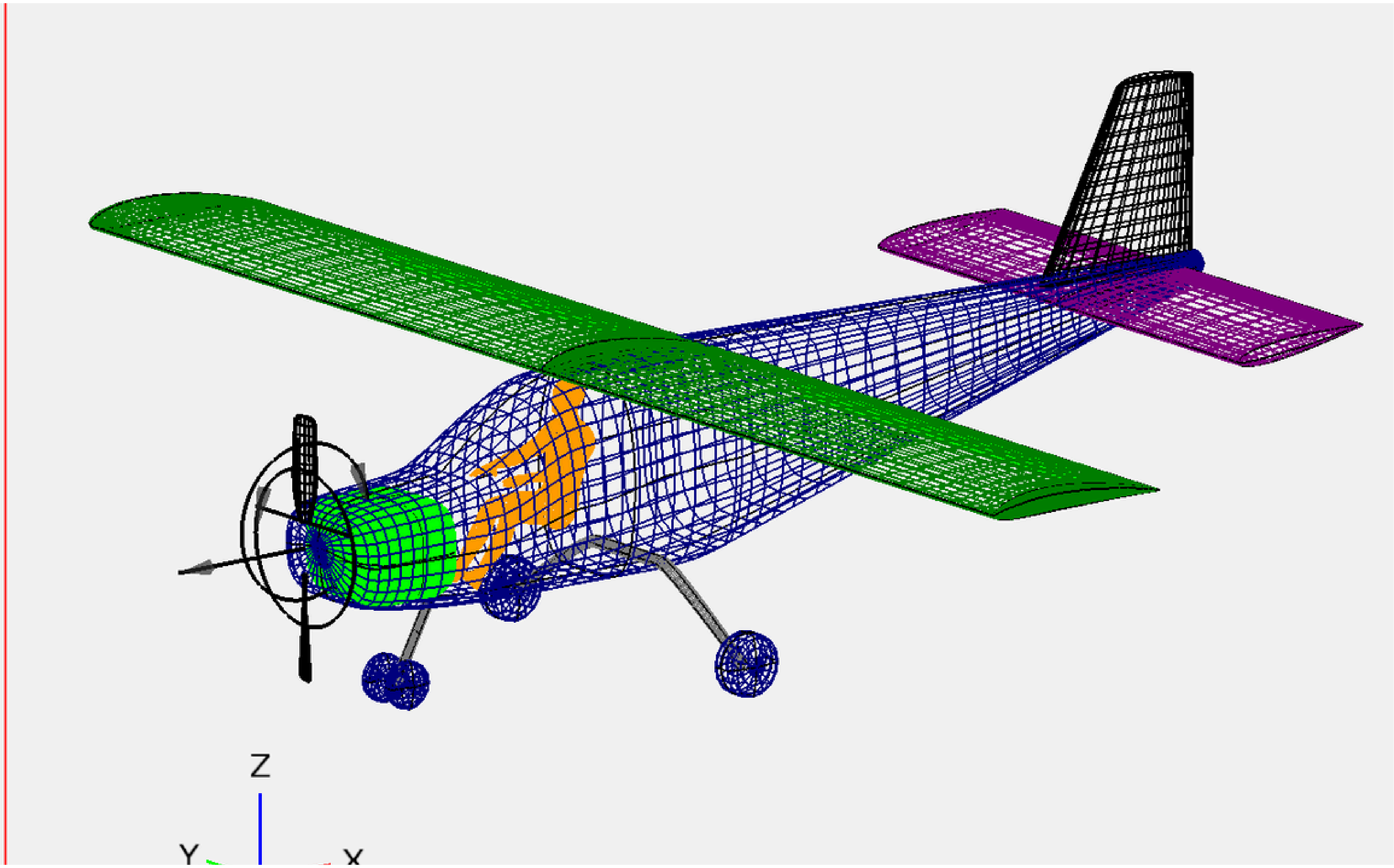
CD vs alpha for vertical stabiliser
0.0125
0.0100
0.0075
0.0050
0.0025
0.0000
CD
0.01102627973
0.01027433657
0.0110262806
-3
-2
-1
0
1
2
3
alpha wrt stabiliser

3.4: Main Observations and Interpretations

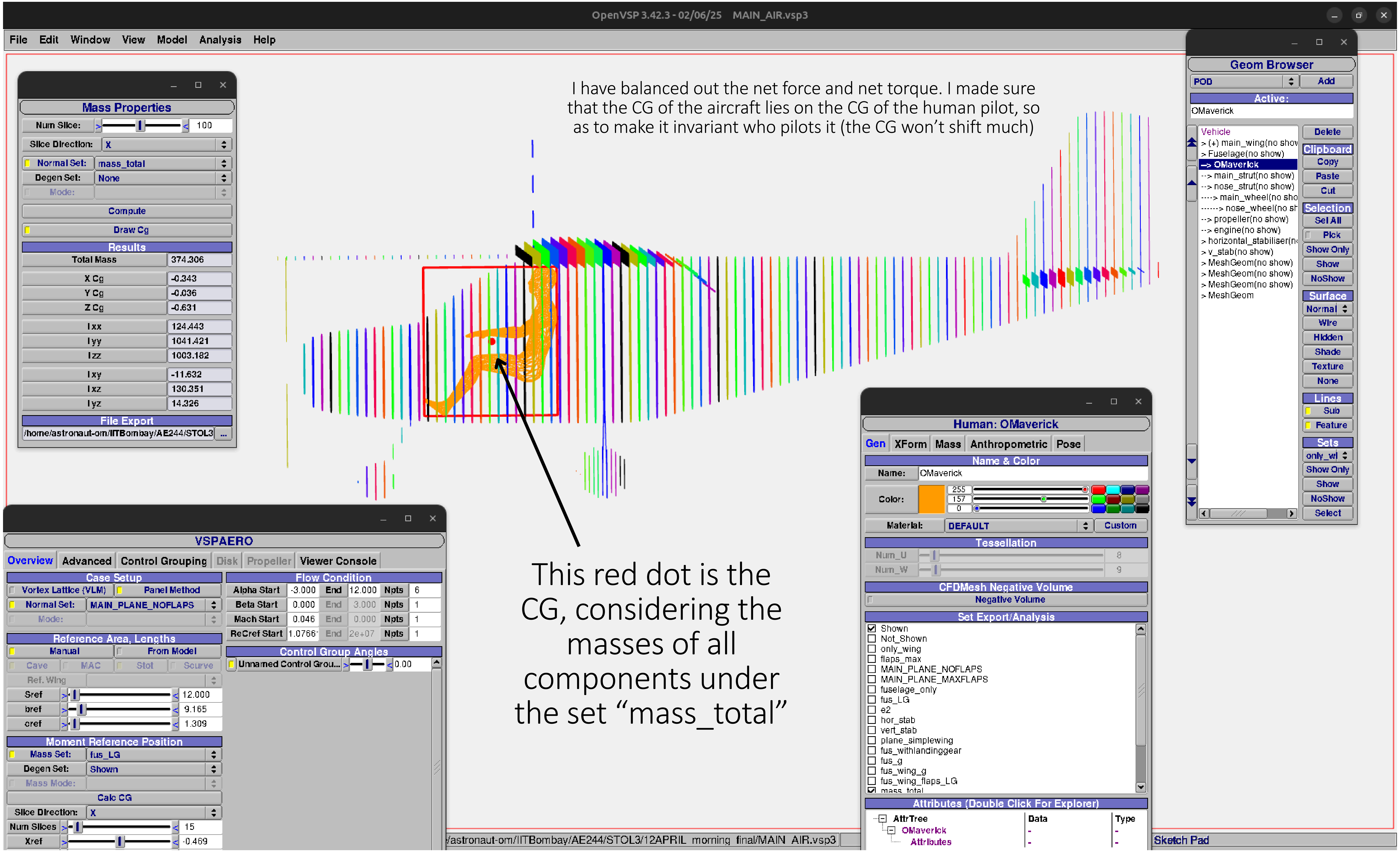
1. CL vs Alpha (Angle of Attack)
   - Horizontal Stabilizer:
     - CL increases linearly with alpha, showing typical lift behavior for symmetric airfoils.
     - Negative alpha results in negative lift (downforce), while positive alpha generates positive lift.
     - At alpha = 0, CL = 0, confirming the symmetric nature of the airfoil.
   - Vertical Stabilizer:
     - Similar linear relationship between CL and alpha.
     - Negative alpha leads to negative lift (yawing moment), while positive alpha generates positive lift.
     - The slope is less steep compared to the horizontal stabilizer, indicating lower lift sensitivity.
2. CD vs Alpha (Angle of Attack)
   - Horizontal Stabilizer:
     - CD shows a parabolic trend with respect to alpha.
     - Minimum drag occurs near alpha = 0, as expected for symmetric airfoils.
     - Drag increases at higher positive and negative angles due to increased flow separation.
   - Vertical Stabilizer:
     - Similar parabolic behavior observed in CD vs alpha.
     - Minimum drag occurs at alpha = 0; drag rises symmetrically with increasing magnitude of alpha.

Section 4:
Overall Aircraft

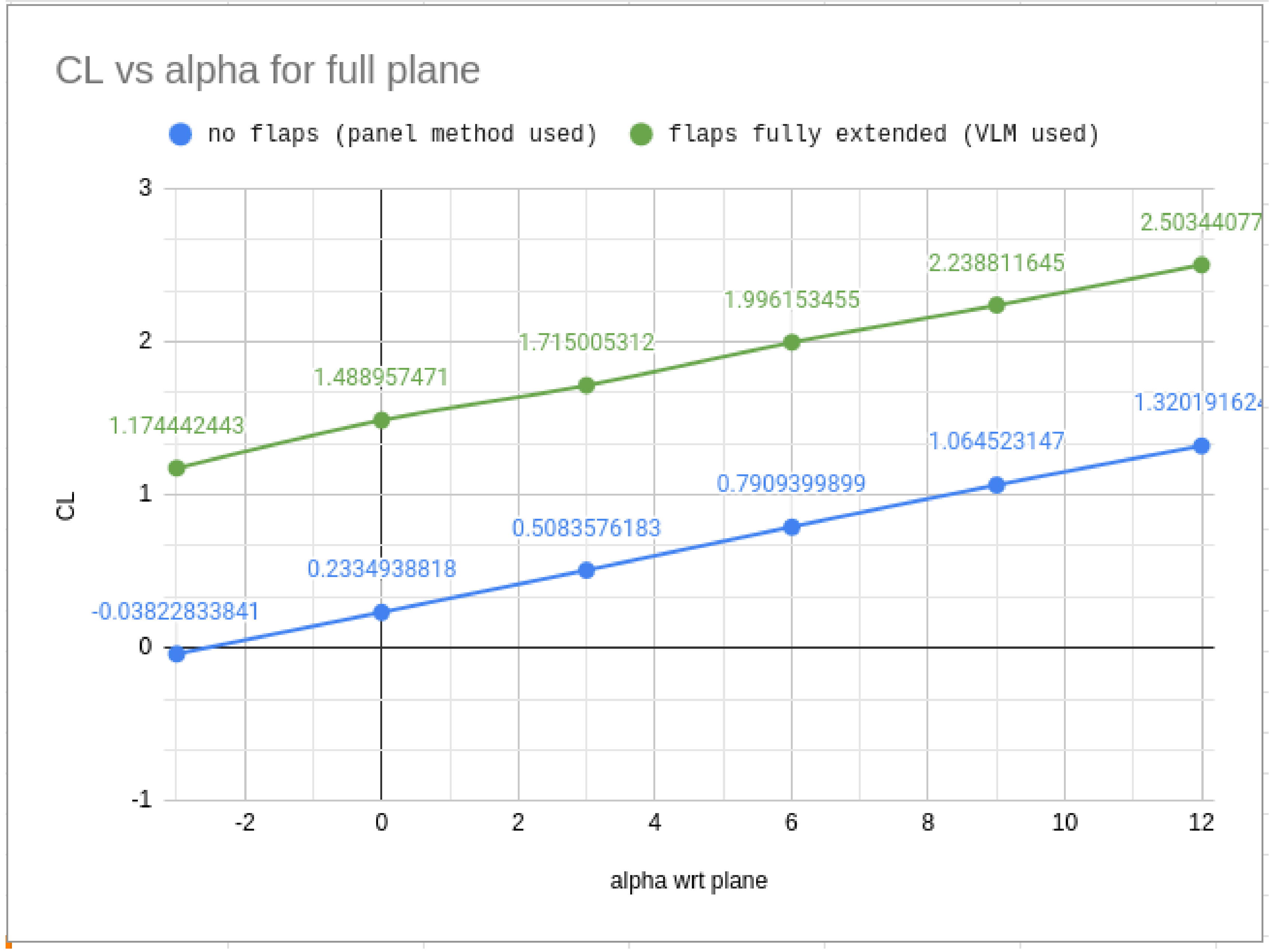
4.1:
Aircraft Design



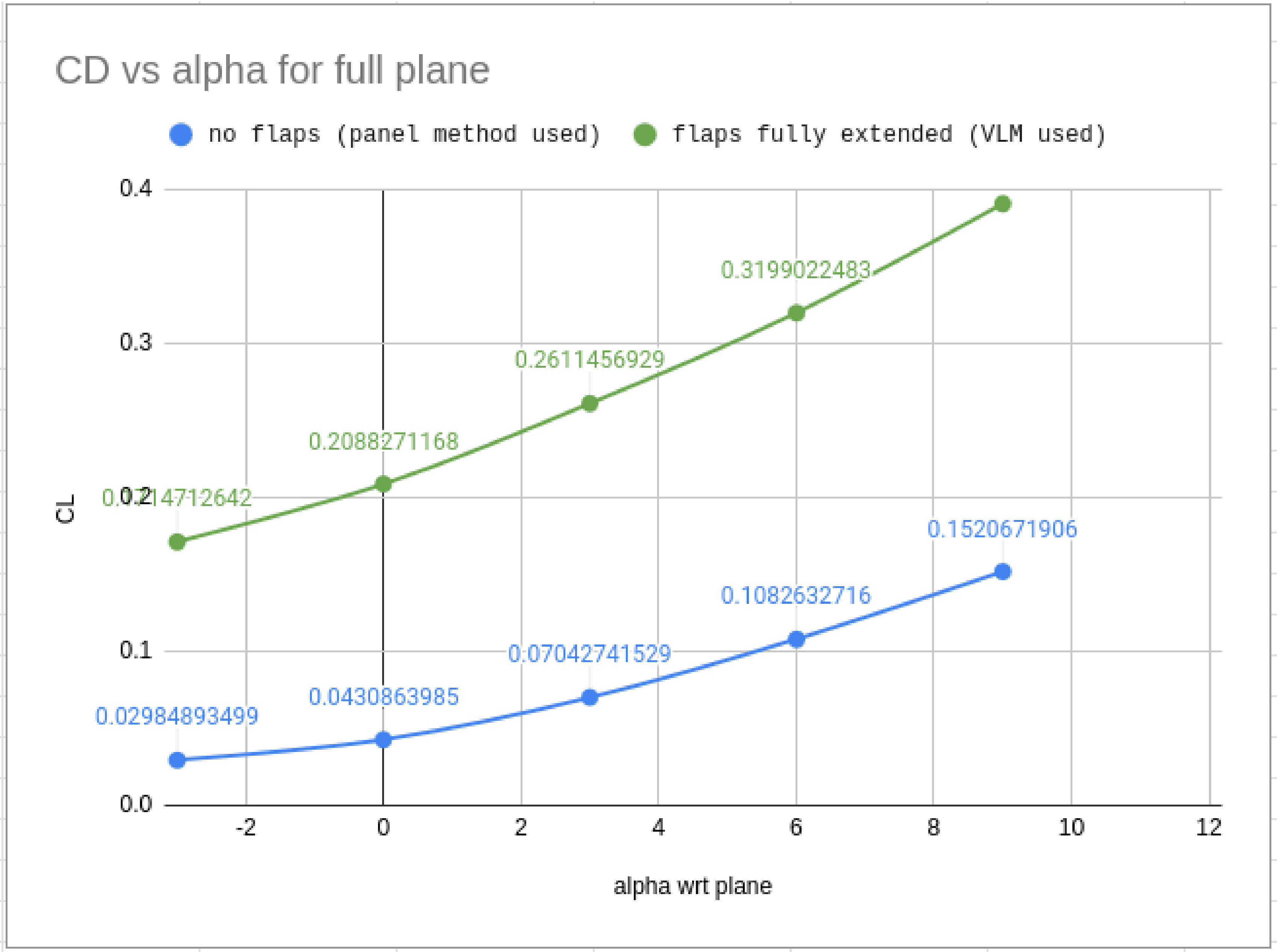
Component	Placement	Rationale
Wing	Highwing configuration	Adds stability since i am no adding any dihedral. Also, its easier for the pilot (me) to see the ground while landing
Fuselage	Central body	To house the engine and pilot.
Horizontal Stabilizer	Rear of fuselage, below the vertical stabiliser	Avoided the T-tail configuration since STOl aricrafts need to operate at high angles of attack and at such a high AoA, T-tails become dysfunctional if there is a stall.
Vertical Stabilizer	Rear of the fuselage	No other location. mounting it at the front will destabilise the plane.
Engine and Propeller	Front of fuselage	We want the CG to be in front of wing, and since engine is the heaviest component, placing it at rear will necessitate a longer and thus heavier fuselage, and so it has been placed at the front. Also, some propwash passing through the wings will aid in the lift as well.
Landing Gear	tricycle configuration	• Provides a stable ground roll because main wheels are behind the centre of mass. • While braking, there is no possibility of a tip over. This possibility exists in taildraggers and it can damage the propeller. Thus we can apply more amount of brakes and land in shorter distances. • In a taildragger configuration, the pilot seats in an awkward postion. Also at higher angles of attack while landing, it will be easier to see the runway if we use a tricycle landing configuration.
Flaps	Towards the root of the wing (the light green part seen here hidden by the wings)	When engaged, some part of the flaps will be in the propwash, generating a greater amount of lift.


OpenVSP 3.42.3 - 02/06/25 MAIN_AIR.vsp3
File Edit Window View Model Analysis Help
I have balanced out the net force and net torque. I made sure that the CG of the aircraft lies on the CG of the human pilot, so as to make it invariant who pilots it (the CG won't shift much)
This red dot is the CG, considering the masses of all components under the set "mass_total"
Mass Properties
Num Slice: 100
Slice Direction: X
Normal Set: mass_total
Degen Set: None
Mode:
Compute
Draw Cg
Results
Total Mass 374.306
X Cg -0.343
Y Cg -0.036
Z Cg -0.631
Ixx 124.443
Iyy 1041.421
Izz 1003.182
Ixy -11.632
Ixz 130.351
Iyz 14.326
File Export
/home/astronaut-om/IITBombay/AE244/STOL3
VSPAERO
Overview Advanced Control Grouping Disk Propeller Viewer Console
Case Setup
Vortex Lattice (VLM) Panel Method
Normal Set: MAIN_PLANE_NOFLAPS
Mode:
Flow Condition
Alpha Start -3.000 End 12.000 Npts 6
Beta Start 0.000 End 3.000 Npts 1
Mach Start 0.046 End 0.000 Npts 1
ReCref Start 1.0766 End 2e+07 Npts 1
Reference Area, Lengths
Manual From Model
Cave MAC Stot Scurve
Ref. Wing
Sref 12.000
bref 9.165
cref 1.309
Control Group Angles
Unnamed Control Grou... 0.00
Moment Reference Position
Mass Set: fus_LG
Degen Set: Shown
Mass Mode:
Calc CG
Slice Direction: X
Num Slices 15
Xref -0.469
Human: OMaverick
Gen XForm Mass Anthropometric Pose
Name & Color
Name: OMaverick
Color: 255 157 0
Material: DEFAULT Custom
Tessellation
Num_U 8
Num_W 9
CFDMesh Negative Volume
Negative Volume
Set Export/Analysis
Shown
Not_Shown
only_wing
flaps_max
MAIN_PLANE_NOFLAPS
MAIN_PLANE_MAXFLAPS
fuselage_only
fus_LG
e2
hor_stab
vert_stab
plane_simplewing
fus_withlandinggear
fus_g
fus_wing_g
fus_wing_flaps_LG
mass_total
Attributes (Double Click For Explorer)
AttrTree Data Type
OMaverick
Attributes
Geom Browser
POD Add
Active:
OMaverick
Vehicle
> (+) main_wing(no show)
> Fuselage(no show)
-> OMaverick
--> main_strut(no show)
--> nose_strut(no show)
----> main_wheel(no show)
------> nose_wheel(no show)
--> propeller(no show)
--> engine(no show)
> horizontal_stabiliser(no show)
> v_stab(no show)
> MeshGeom(no show)
> MeshGeom(no show)
> MeshGeom(no show)
> MeshGeom
Delete Clipboard Copy Paste Cut
Selection Sel All Pick Show Only Show NoShow
Surface Normal Wire Hidden Shade Texture None
Lines Sub Feature
Sets only_wi Show Only Show NoShow Select
/astronaut-om/IITBombay/AE244/STOL3/12APRIL_morning_final/MAIN_AIR.vsp3
Sketch Pad

4.2:
CL & CD vs α

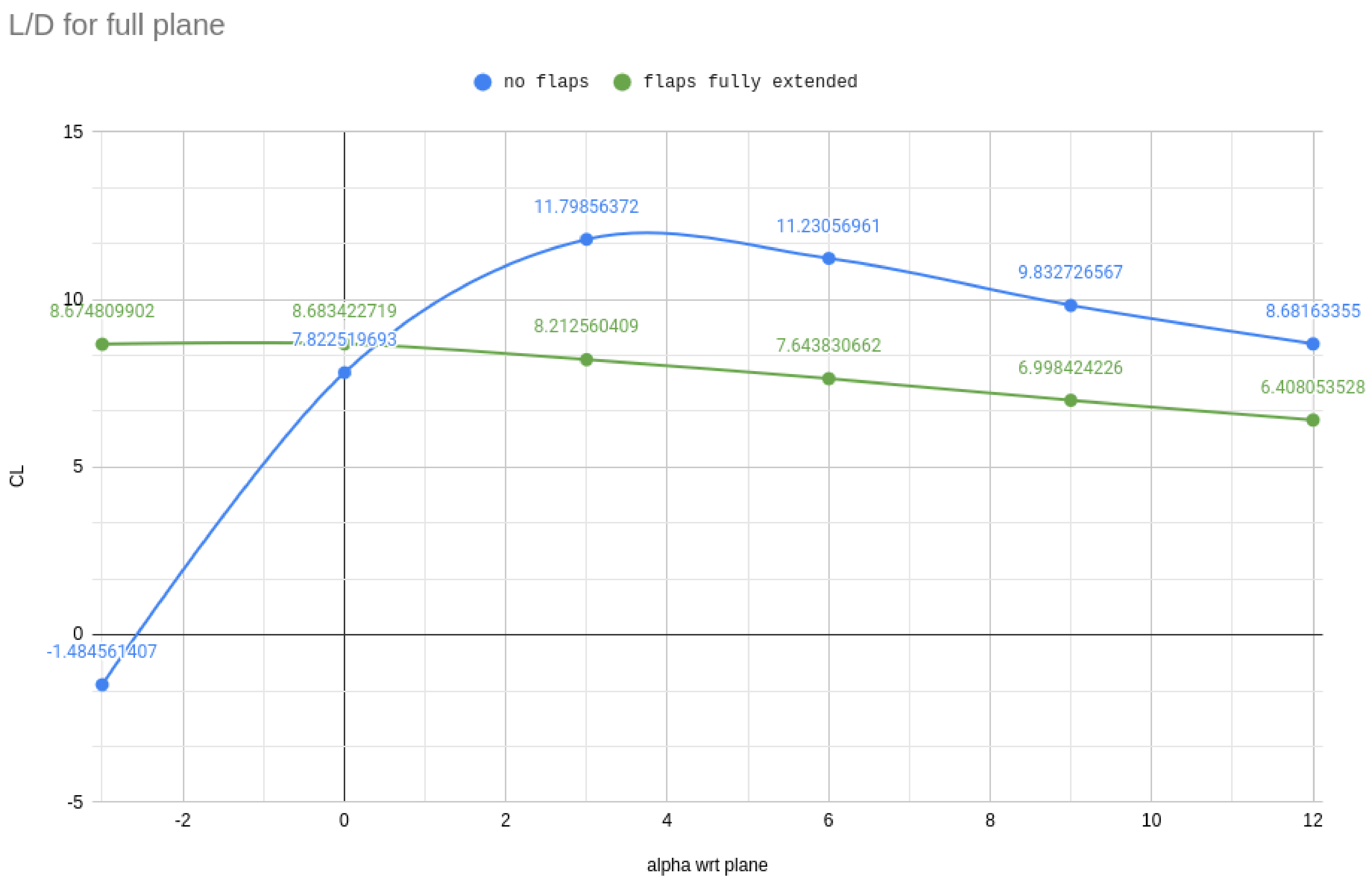


CL vs alpha for full plane
no flaps (panel method used)
flaps fully extended (VLM used)
CL
alpha wrt plane
1.174442443
1.488957471
1.715005312
1.996153455
2.238811645
2.50344077
-0.03822833841
0.2334938818
0.5083576183
0.7909399899
1.064523147
1.32019162



CD vs alpha for full plane
no flaps (panel method used)
flaps fully extended (VLM used)
CL
alpha wrt plane
0.2088271168
0.2611456929
0.3199022483
0.02984893499
0.0430863985
0.07042741529
0.1082632716
0.1520671906

L/D vs α



L/D for full plane
no flaps
flaps fully extended
15
10
5
0
-5
CL
-2
0
2
4
6
8
10
12
alpha wrt plane
11.79856372
11.23056961
9.832726567
8.68163355
8.674809902
8.683422719
7.822519693
8.212560409
7.643830662
6.998424226
6.408053528
-1.484561407

4.3: Weight Estimates

Component	Weight Estimate	Method of Estimation
Wing	38.4 kg	Code-based estimation using carbon fiber composite (CFRC). please look into the code.
Fuselage	56.3 kg	same as above
Horizontal Stabilizer	7.7 kg	same
Vertical Stabilizer	5.8 kg	same
Engine and Propeller	(111 + 9)kg	Based on engine manual and Whirwind propeller manual
Fuel + Tank	22.9 kg + 1 kg	Estimated using Breguet Range Equation for a range of 100 km
Cockpit + Controls	40 kg	Searched using chatgpt. Its generally 15 to 30 kg. Added 10 kg margin here.
Passenger	51 kg	I weighed myself before filling this
Landing gear	20 kg	it is generally ~5% of Takeoff mass (0.05*373 = 18.6 kg)(2 kg margin added)
Payload	10 kg	(this can be used as a margin too)
Total	373.1 kg	*Total Margin = 22 kg. Also this is for 100 km, so reducing it to 10 km will increase the margin by 4 kg. So 26 kg of margin.*

Section 5:

Design Validation

5.1
Aircraft Performance Estimation

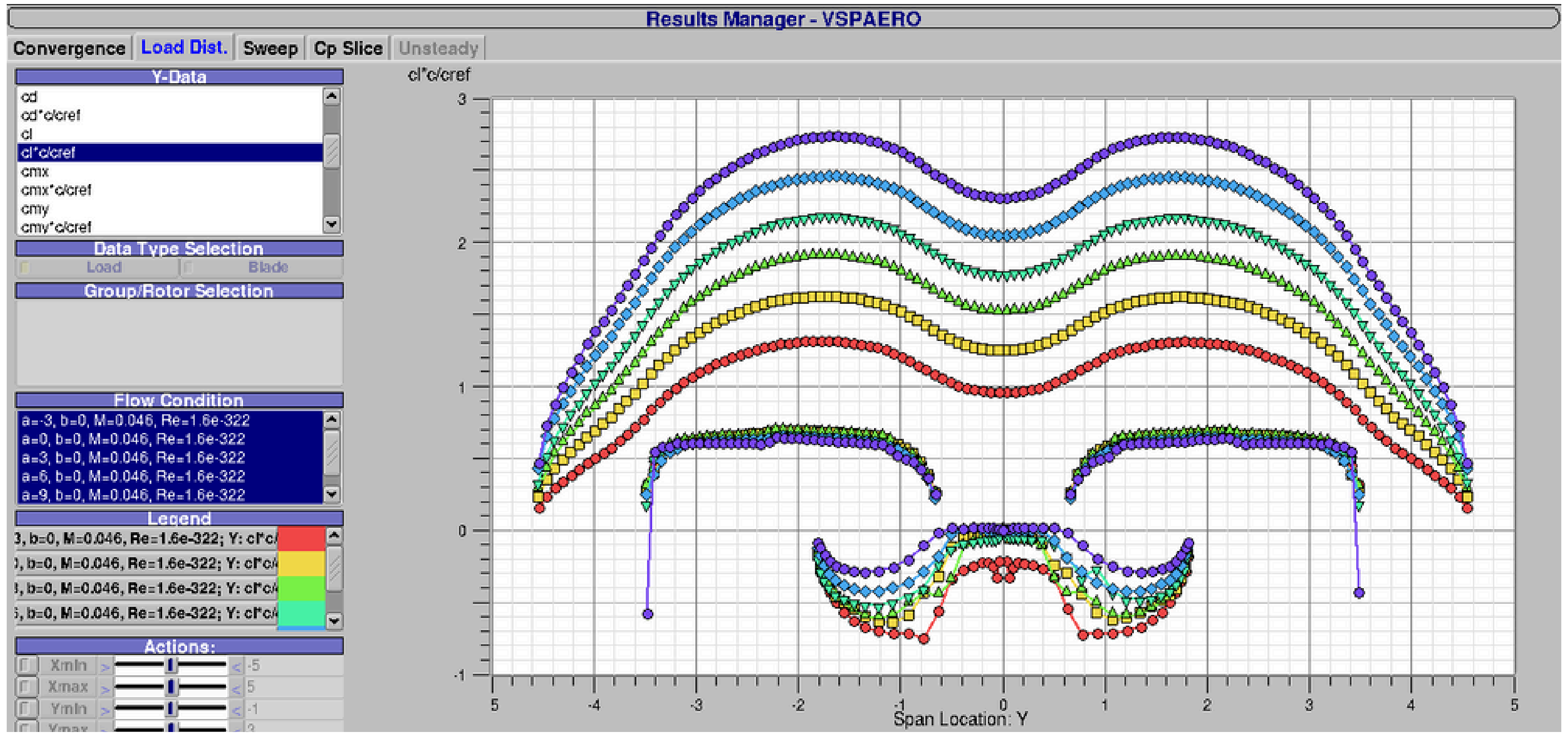
Parameter	Value	Method of Estimation
Takeoff Speed	15.60 m/s	code. I also consider the effect of thrust reduction as the aircraft moves forward
Takeoff Distance	18.98 m (with flaps)	code. iteratively calculated distance using thrust using python code.
Maximum Speed	64 m/s	google sheet. I balanced out the drag to thrust. also considered the reduction in thrust as aircraft speed increases, by trial and error and found this speed looks good.
Maximum Endurance Speed	52 m/s	finding the minimum value of (Cd^1.5 / CL^0.5). since there are not enough data points for values less than alpha - 0 deg. and it was found that minimum of this ratio occurs at AoA = 0. and out velocity cruise was found out to be 52 m/s

5.2
Comments on Aircraft Performance

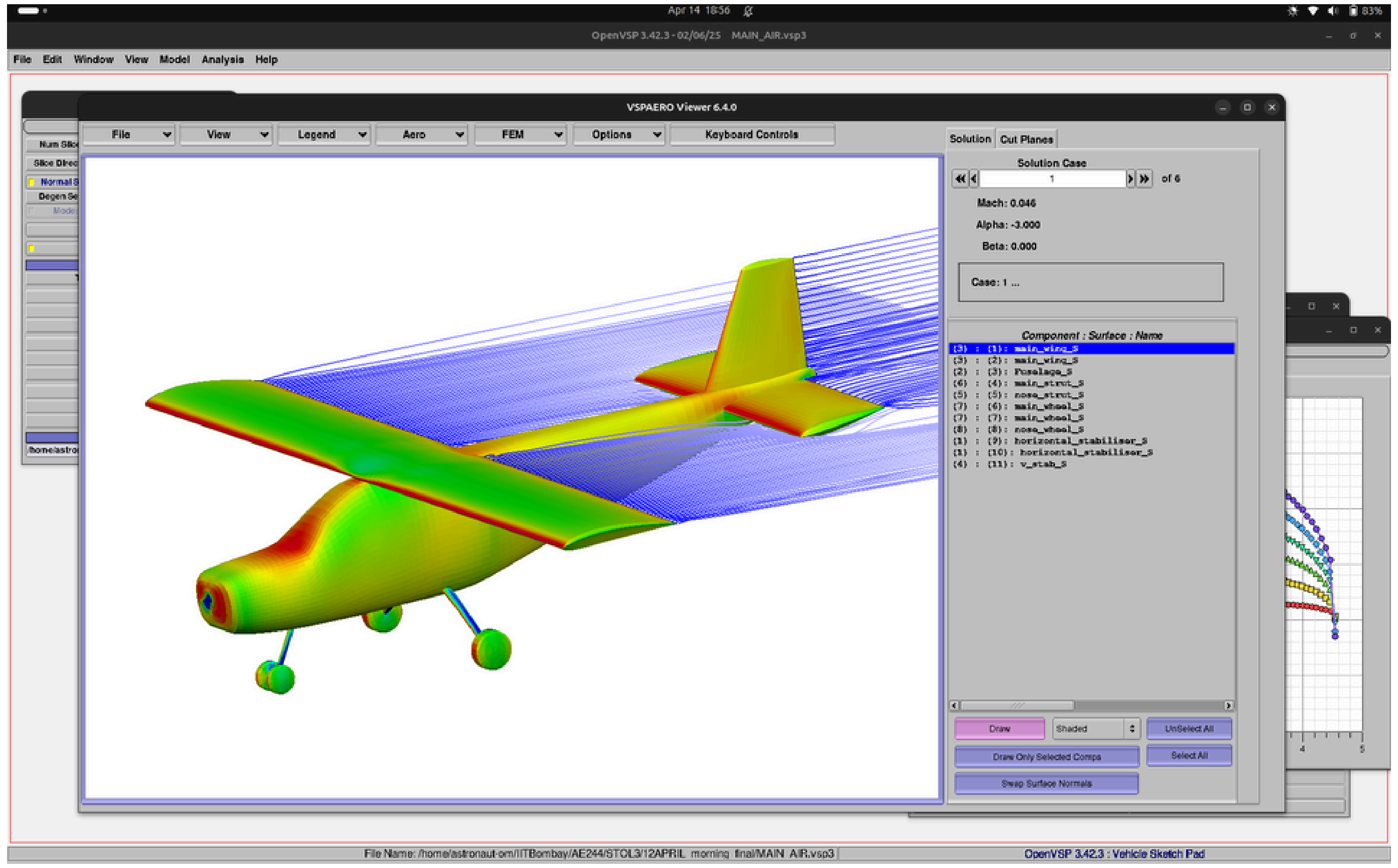
- max takeoff location altitude: 806 m above Mean Sea Level
- maximum takeoff location temperature : 50 °C
- range when operated at max altitude and pressure with no payload: 100 km
- I have made the aircraft out of carbon fibre fully. This will be costly endeavour. The plane can be made of fibreglass as well and we can come up with the numbers using the code.
- The aircraft is stable during cruise as i have balance out the net forces and torque. This can be seen using Cm graphs as well.
- The tail sizing is sufficient to provide the moment required to pitch up during takeoff
- It was assumed that the aircraft won’t go above 500m from ground. This is to simplify the design process.
- Given the margin for mass kept, the range of 100 km, and the maximum temperature it can take off at is 50 °C this aircraft will surely take off within 20 m and a range of > 8km. Thus this aircraft satisfies the original requirement given.
- The performance can be improved using a better lightweight engine.
- Electric planes can have a higher power to weight ratio than piston engine. Since we require a range of only 8 km, it makes sense to switch to electric propulsion for this plane.
- The vertical stabiliser can be mounted at an angle to counteract the torque due to the use of a single engine.

Section 6:
Detailed Design and Extreme STOL Capability (Bonus)

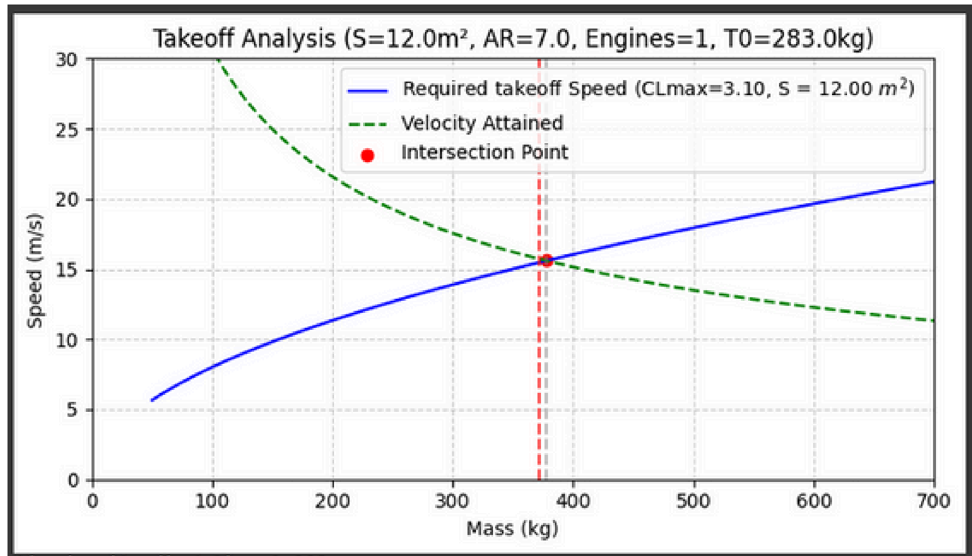
6.1
Detailed Design



wing loading



6.1 Detailed Design

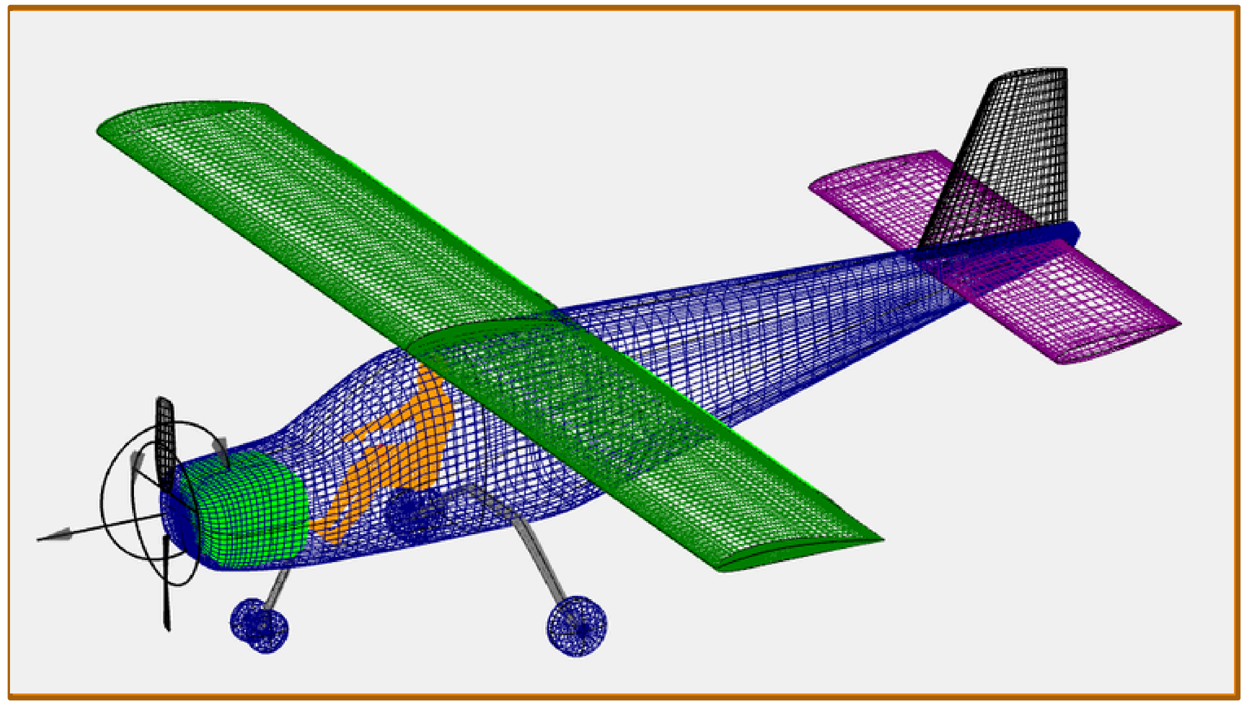



```
Air density (kg/m³):  0.992
Air pressure (bar):   0.920
✅ Good: Total mass is less than or equal to the allowable mass!

--- Performance Metrics ---
Intersection Mass (Max Allowed):  378.1 kg
Takeoff distance:                 18.75 m
Speed at Takeoff:                 15.47 m/s
Lift at Takeoff:                  4413.0 N
Required CL at takeoff:           2.562
Takeoff Mach number:              0.045
Takeoff Reynolds number:          1.1e+06
Required Lift at cruise:          3647.1 N
Required CL at cruise:            0.227
Cruise Mach number:               0.1516
Reynolds number at cruise:        3.6e+06

--- Geometry ---
Wingspan:                    9.1652 m
Chord Length:                1.3093 m
Wing Aspect Ratio:           7.0000
Tail distance:               3.5000 m
Area of horizontal tail:     3.591 m²
Horizontal tail span:        3.545 m
Area of vertical tail:       0.943 m²
Vertical tail span:          1.153 m

--- Mass Breakdown ---
• Wing Mass:              38.4 kg
• Fuselage Mass:          56.3 kg
• Horizontal Tail Mass:   7.7 kg
• Vertical Tail Mass:     5.8 kg
• Engine Mass (1x):     111.0 kg
• Propeller Mass:         9.0 kg
• Controls Mass:          40.0 kg
• Pilot Mass:             51.0 kg
• Payload Mass:           10.0 kg
• Landing Gear Mass:      20.0 kg
• Fuel Mass:              22.6 kg

 Empty Mass of aircraft:   288.2 kg
 Total Takeoff Mass:       371.8 kg
 Allowable Mass:           378.1 kg
 Margin for Mass:          6.30 kg

 Empty weight fraction:    0.78
 Thrust-to-weight ratio:   0.76
```

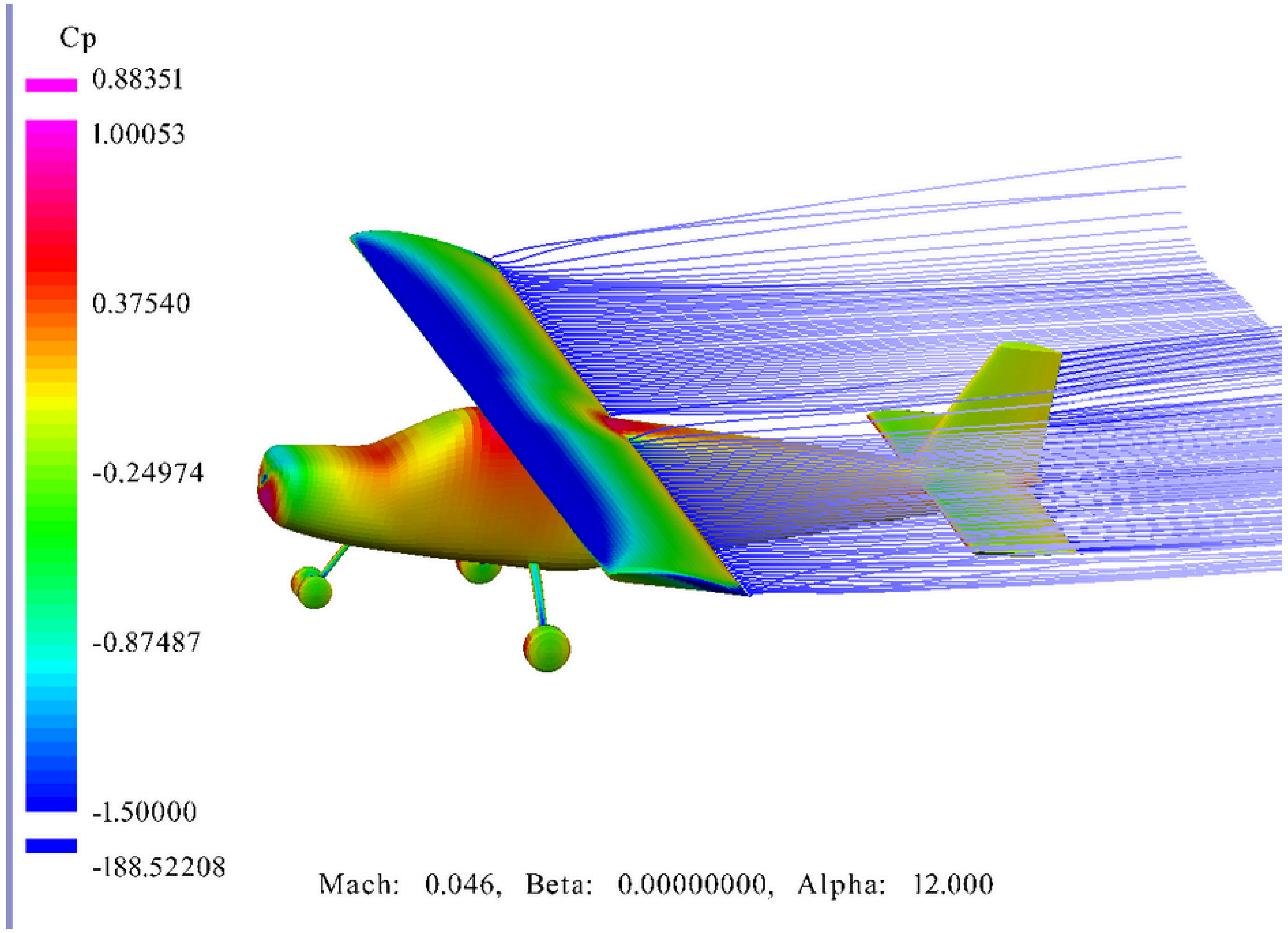
INITIALIZE	1.00
AirP(bar)	0.92
AirTemp(°C)	50.00
Wing Area ...	12.00
Aspect Ratio	7.00
Engines	1
Thrust/Eng...	283.00
Approx. Ta...	17.00
Rolling μ	0.02
CLmax(Fo...	3.10
L/D Max	11.00
Cruise Spe...	52.00
Range (km)	100.00
tail_dist(m)	3.50
Payload(kg)	10.00
Wing Mate...	Carbon Fibre
Fuselage M...	Carbon Fibre
Horiz Tail ...	Carbon Fibre
Vert Tail Mat.	Carbon Fibre


```
--- Mass Breakdown ---
• Wing Mass:              38.4 kg
• Fuselage Mass:          56.3 kg
• Horizontal Tail Mass:   7.7 kg
• Vertical Tail Mass:     5.8 kg
• Engine Mass (1x):     111.0 kg
• Propeller Mass:         9.0 kg
• Controls Mass:          40.0 kg
• Pilot Mass:             51.0 kg
• Payload Mass:           10.0 kg
• Landing Gear Mass:      20.0 kg
• Fuel Mass:              22.6 kg

 Empty Mass of aircraft:   288.2 kg
 Total Takeoff Mass:       371.8 kg
 Allowable Mass:           378.1 kg
 Margin for Mass:          6.30 kg

 Empty weight fraction:    0.78
 Thrust-to-weight ratio:   0.76
```

6.1
Detailed Design

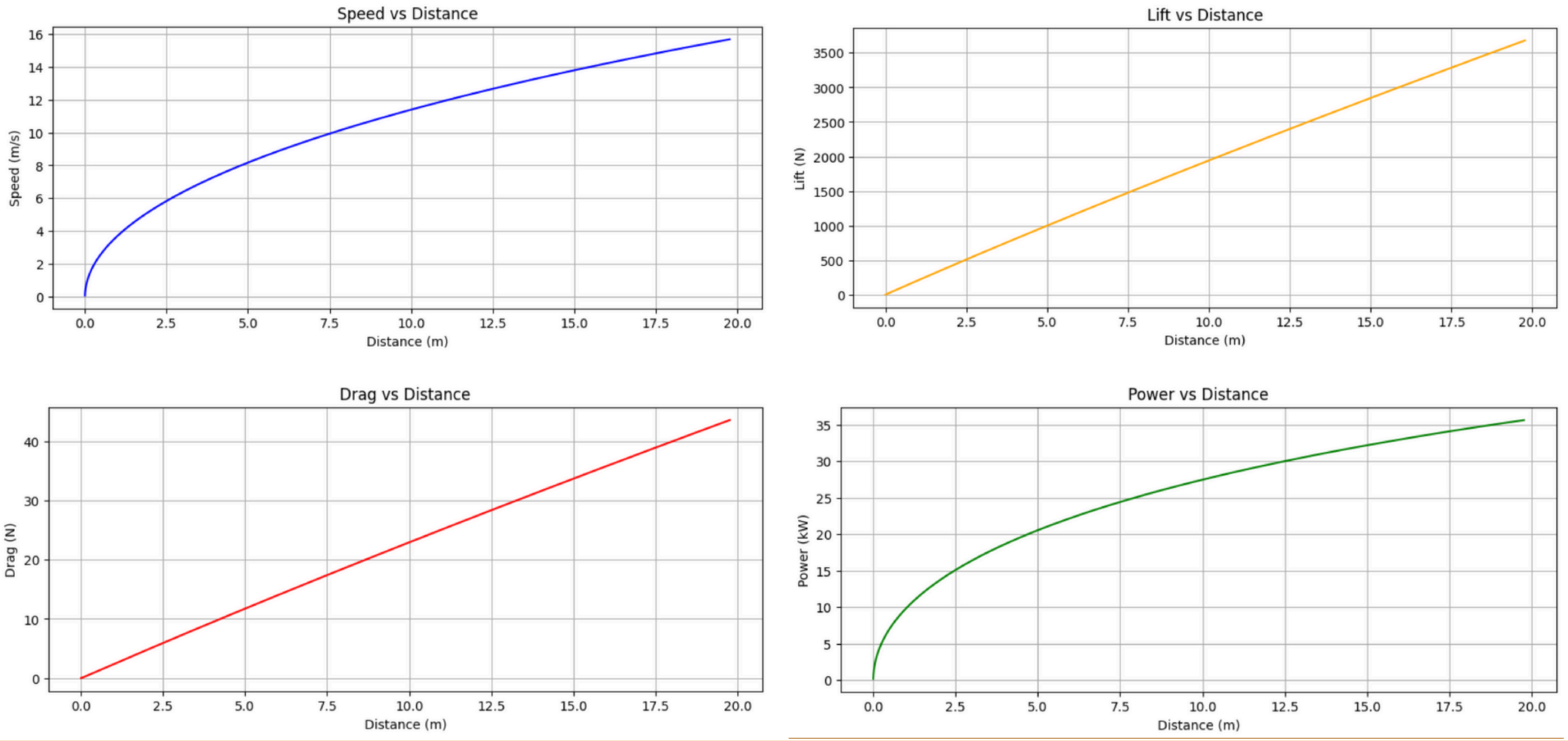


- Lift at take-off conditions: 4413 N
- Drag at take-off conditions: 687 N
- alpha = 12 degree



Cp contour

6.2
Take-off Performance



Acknowledgement

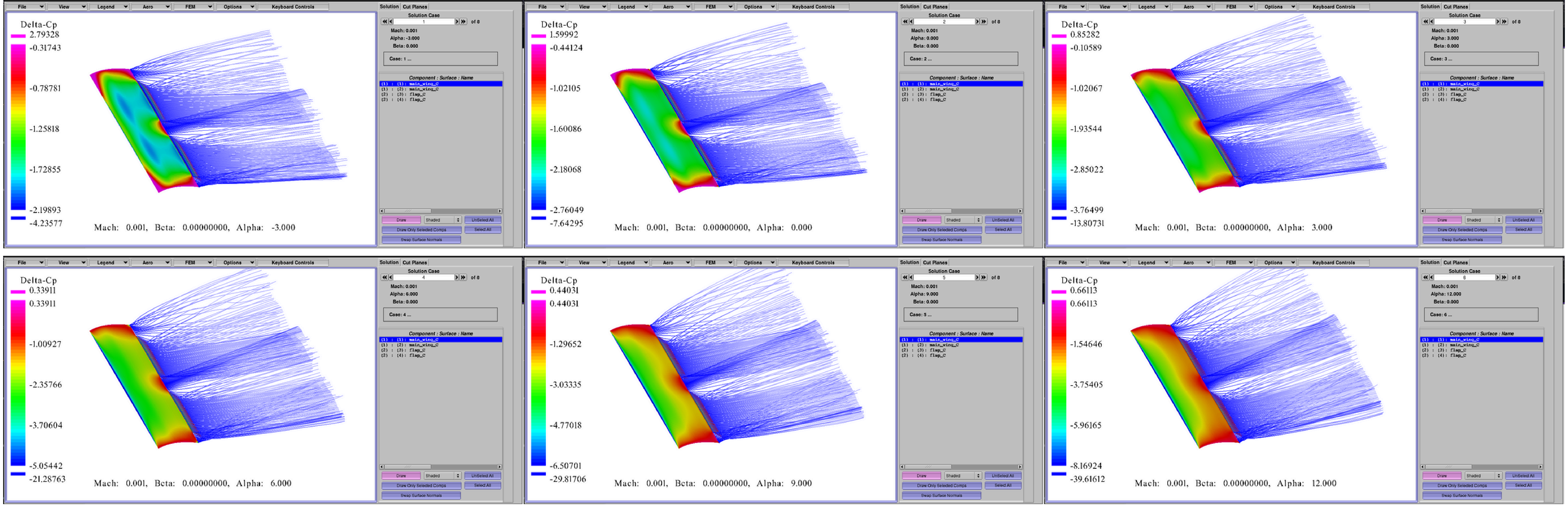
- Samanth Martis 23b0046: Lifting line theory code

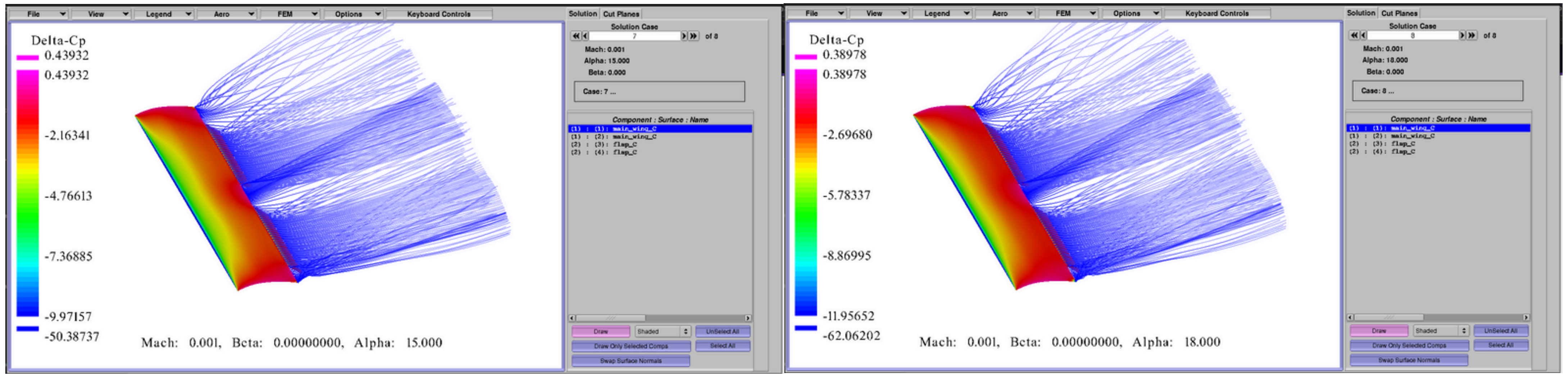
References

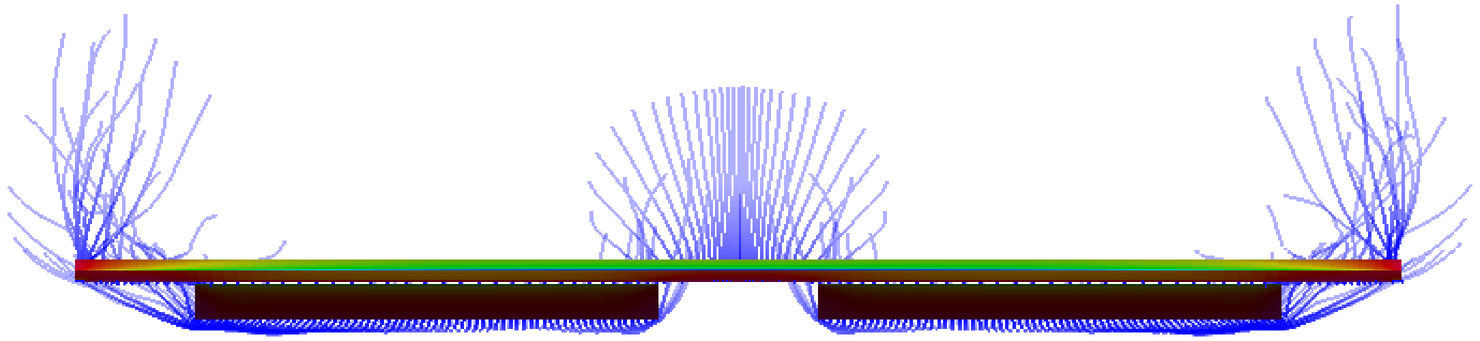
- < List all references (books, paper, websites, etc.) used while doing the assignment>
- Aircraft Design by Daniel P. Raymer
- https://zenithair.net/introduction-stol-ch750/
- https://en.wikipedia.org/wiki/STOL
- https://www.lycoming.com/sites/default/files/attachments/O-320%2520Operator%2520Manual%252060297-30.pdf
- https://www.mide.com/air-pressure-at-altitude-calculator
- https://aero.us.es/adesign/Slides/Extra/Stability/Design_Tail/Chapter%206.%20Tail%20Design.pdf
- https://aviation.stackexchange.com/questions/35069/how-do-hoerner-wingtips-work
- Thrust: https://vansairforce.net/threads/static-thrust-testing.113438/
- Propeller: https://whirlwindpropellers.com/aircraft/product/ga200l-2-blade-propeller-for-lycoming-o-320-io-360/
- chatgpt
- perplexity AI

I learnt a lot about aircraft design through this and ended up making a code for designing an STOL aircraft. OpenVSP was a bit difficult to tame and sometimes didn’t give proper results, but overall its good for modelling an aircraft. It was a rewarding experience

THANK YOU







At takeoff, trailing edge vortex distribution