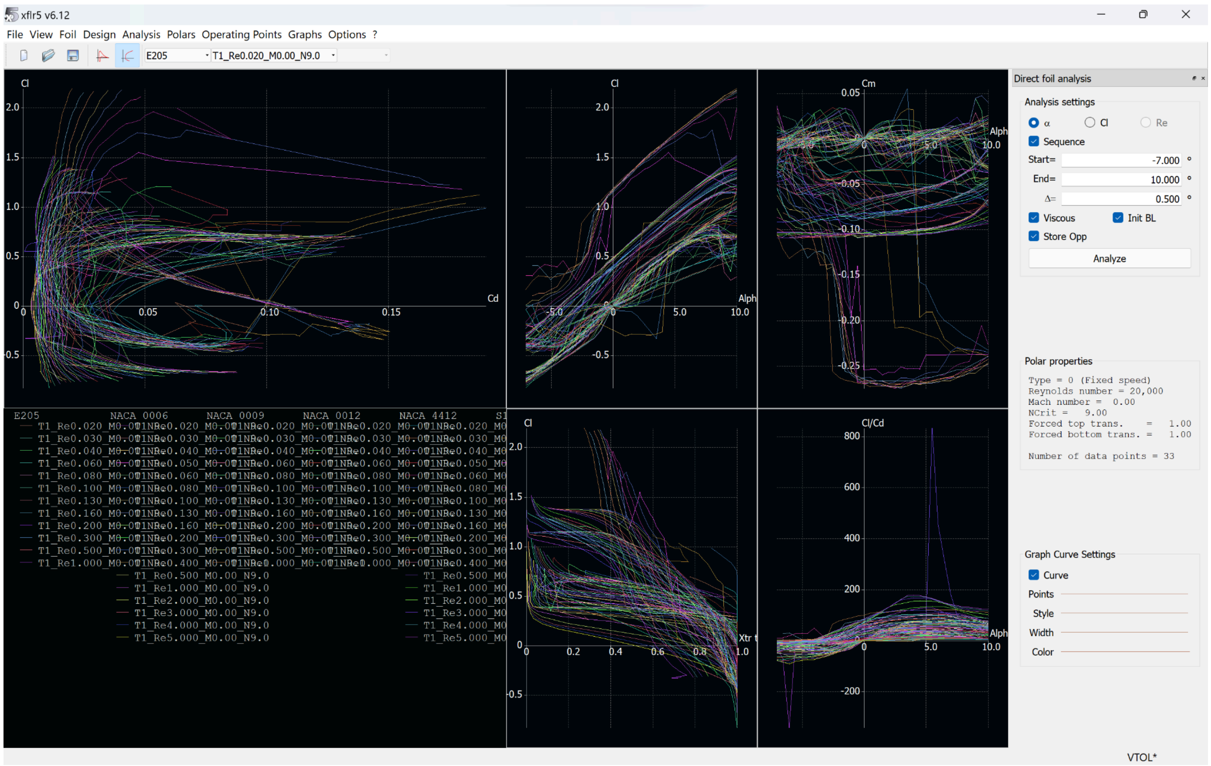
XFLR5 Report

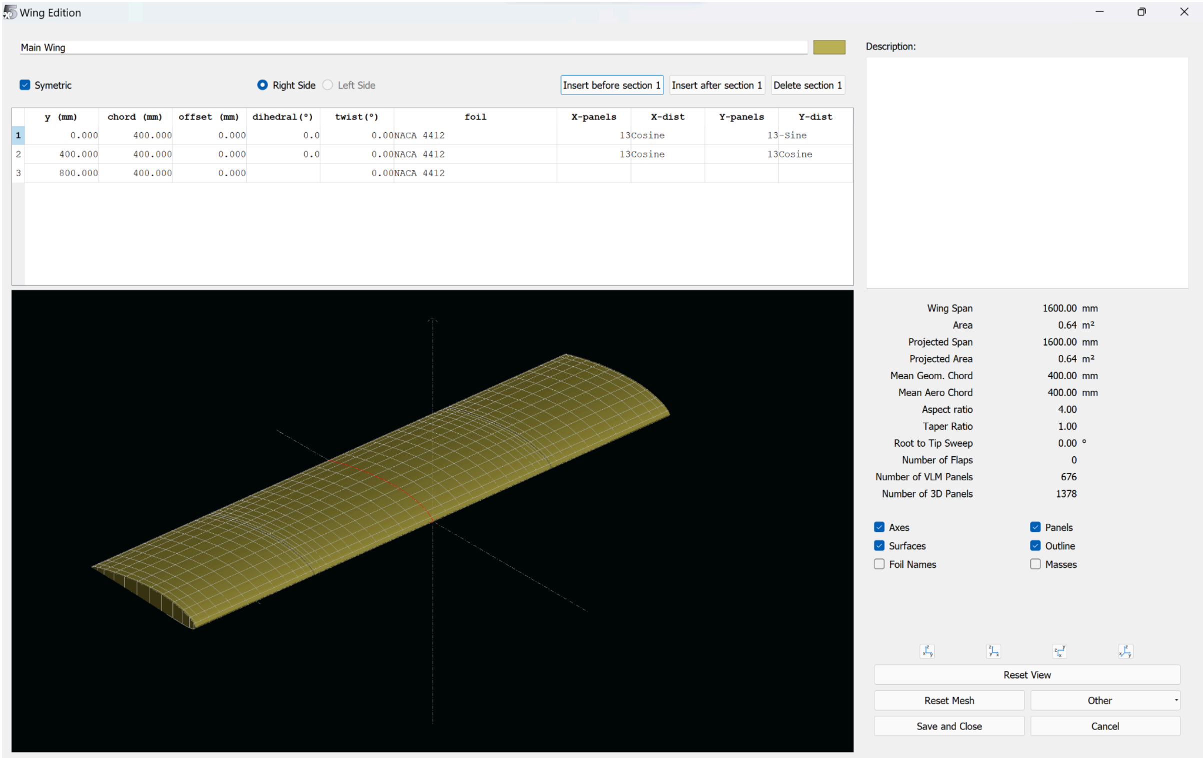
Airfoil Selection:-

1) We have analyzed various prospective airfoils for our VTOL design in XFLR5 software which implements the Vortex Lattice Method along with the use of panels to interpolate the 2D results it calculates using a mathematical model into 3D analysis for airfoils.
2) We have performed multi-threaded batch analysis from -7 to 10 degrees with a change of 0.5 degrees.



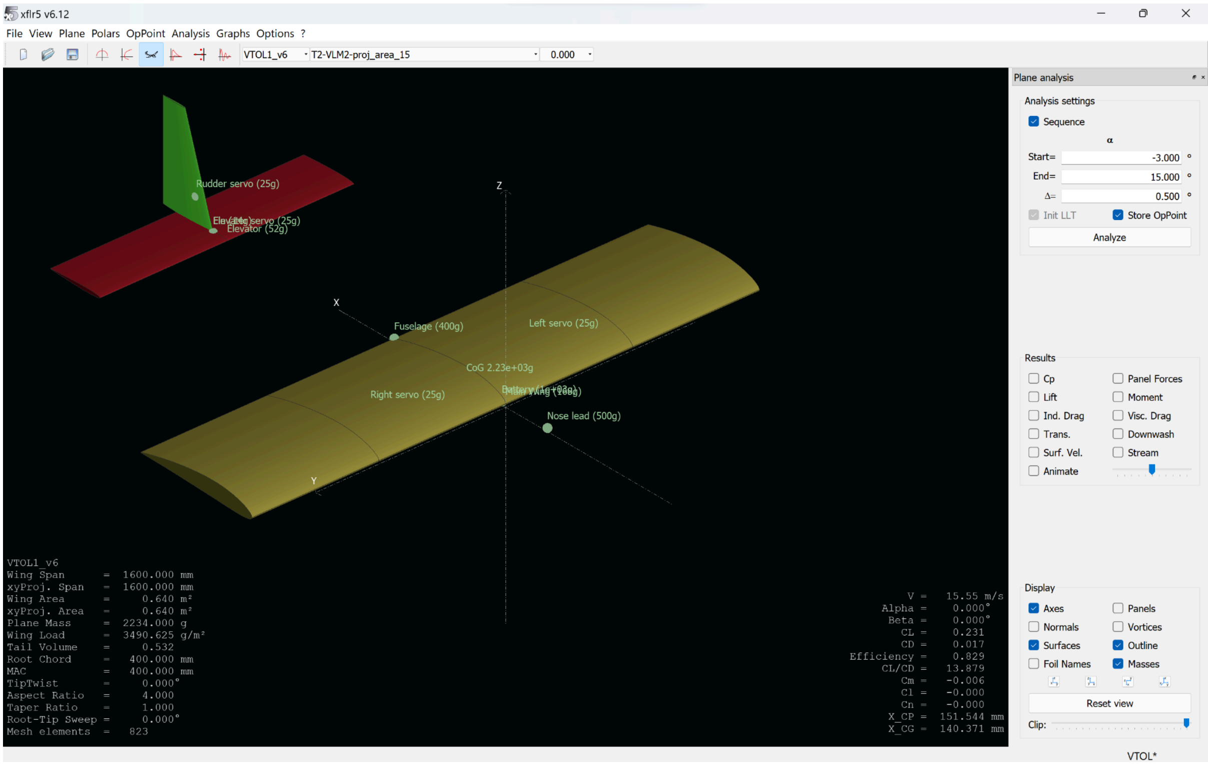
3) We have identified transitional behavior in some of the foils, which we have investigated using Type 4 Analysis. It shows the variation of Cl and Cd with Re. For improved performance at lower Re, we have changed the forced transition points.

4) We have also analyzed the separation bubbles. The least drag was observed for those angles of attack where the bubble was at the trailing edge.
5) We have performed flapped foil from -6 to 10 degrees with a change of 0.25 degrees.
6) On observing some irrational points in the polar graphs, we identified those polars, located the particular operating point, and ran the analysis with new AOA values while remembering to initialize the Boundary Layer as we are using a new AOA range.
7) We have also investigated inverse analysis to get modified airfoils (full/mixed inverse) by specifying the required plots.
8) For wing and plane design, we performed direct analysis from -7 to 10 degrees with a change of 0.5 degrees.XFLR5 interpolates this 2D data into 3D. Sometimes, this convergence fails because XFLR5 doesn’t have the required 2D data to interpolate (“out of envelope” issue). The solution to this issue is explained later.
9) In wing and plane design, we added and defined all the respective sections, and applied panels properly at section boundaries (the number and type of panels are 13 and cosine respectively). We gave a z offset (of 10 mm) to the main wing, fin, and elevator to prevent division by 0. A further offset of 5 mm was given to the servos.

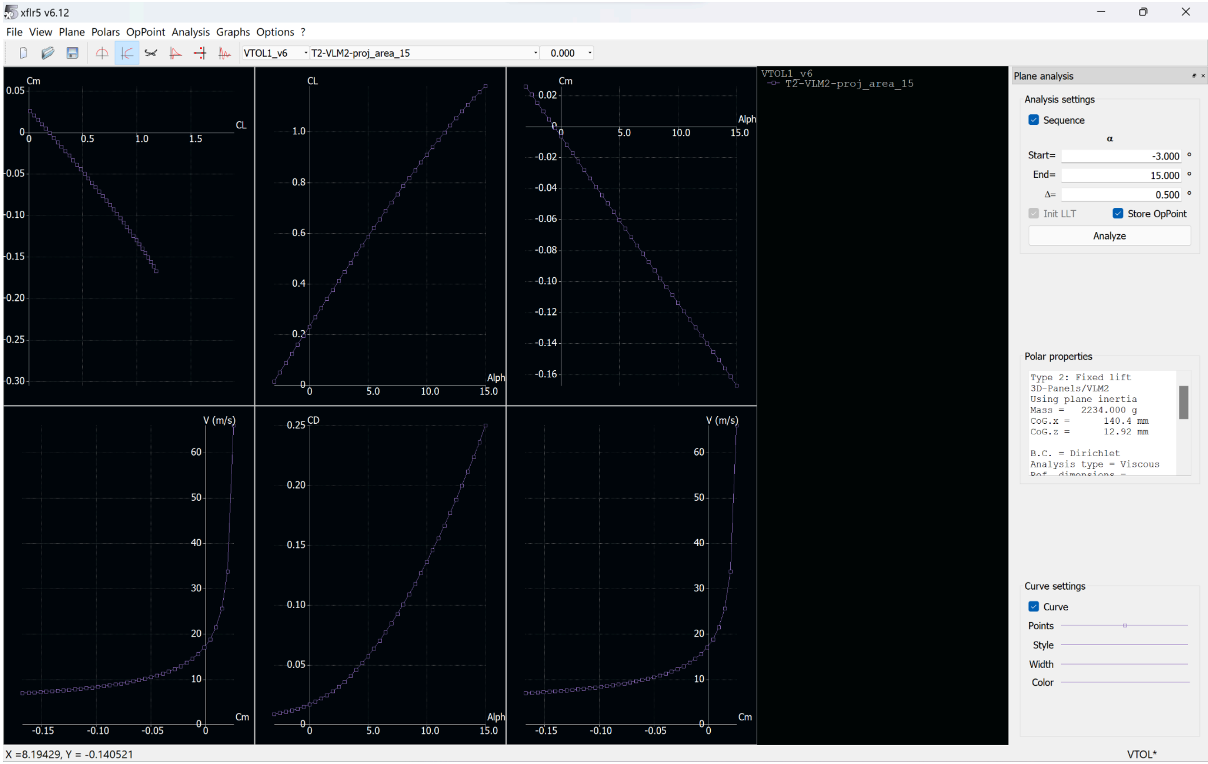


	y (mm)	chord (mm)	offset (mm)	dihedral(°)	twist(°)	foil	X-panels	X-dist	Y-panels	Y-dist
1	0.000	400.000	0.000	0.0	0.00	NACA 4412	13	Cosine	13	-Sine
2	400.000	400.000	0.000	0.0	0.00	NACA 4412	13	Cosine	13	Cosine
3	800.000	400.000	0.000		0.00	NACA 4412				

10) We added all the additional point masses after defining mass distributions in the wing, elevator, and fin.



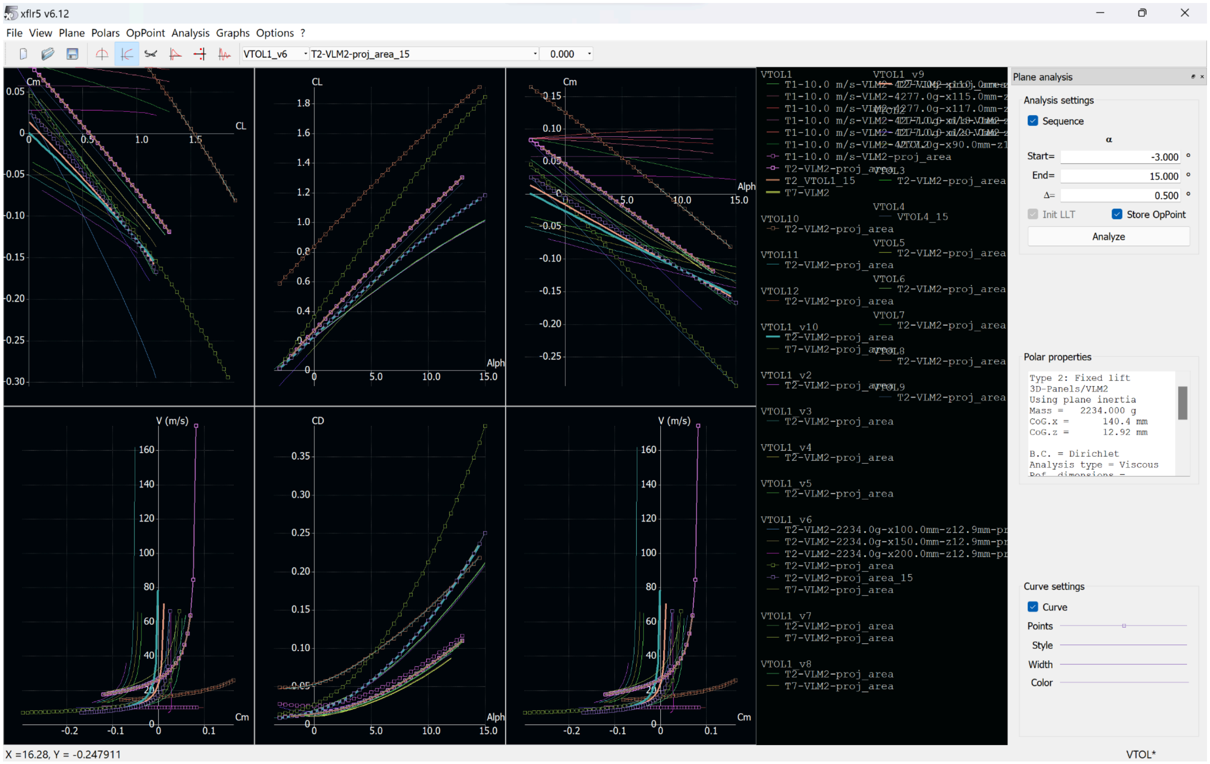
11) Performed Type 2 (cruise) analysis with VLM2 and used plane inertia and wing planform projected on the xy plane.



12) Performed analysis from -3 to 15 degrees with a change of 0.5 degrees. Verified log file for successful analysis.

13) For flapped wings, we inserted two sections at the same location and applied panels properly (only 1 panel at the first of the two intersecting sections).

14) For selecting the final airfoils for the wing, elevator, and fin, we plotted polar curves of all the possible combinations and chose the airfoil combination with the required polar characteristics and L/D ratio.



15) For importing airfoils, we downloaded the txt file from the UIUC airfoil database, arranged it in descending order using Excel, and converted it to a dat file.

16) Viscous analysis errors: XFLR5 uses 2D data from direct analysis and interpolates it into 3D in wing and plane design. To fix “out of the envelope” issues, we checked which Cl values were out of the flight envelope. Then we revisited the direct analysis and increased the AOA range to get those Cl values. Whenever Re values were out of the envelope, we went back to direct analysis and analyzed for suitable Re. As a good practice, we performed direct analysis with a sufficiently large AOA range to begin with. After a certain point, this solution may stop working because the Vortex Lattice Method is linear, and hence, for it, Cl will increase linearly with AOA and it will demand Cl values for interpolation which we just won’t be able to

produce in direct analysis even on enlarging the AOA range because stalling will take place before we can reach that Cl.

17) Stability prerequisites: Analysed Cm vs. AOA and Cm vs. Cl graphs. Both should have a negative slope and Cl should be positive when Cm=0.
18) Neutral Point identification: In the Cm vs. AOA graph, we ran Type 1 Analysis with different CG positions till we found the NP (when Cm is constant).
19) Static Margin = (Xnp - Xcg)/MAC
20) Using the static margin, we redefined the plane inertia and ran a Type 2 analysis. Then we found the L/D ratio, balanced AOA, design Cl, cruise velocity, and other important parameters from the various polar graphs.

Possible Airfoil Combinations

Symmetrical (for elevator and fin)	Asymmetrical (for main wing)
NACA 0006	NACA 4412
NACA 0009	Eppler 205
NACA 0012	MH 114
	SD7037
	NACA 4415
	Eppler 182
	SA7038
	Clark Y
	S1223

Shortlisted Airfoil Combinations

1) NACA 0006 and NACA 4412
2) NACA 0006 and Eppler 205
3) NACA 0006 and SA7038
4) NACA 0009 and NACA 4412
5) NACA 0009 and Eppler 205
6) NACA 0009 and SA7038
7) NACA 0012 and NACA 4412
8) NACA 0012 and Eppler 205
9) NACA 0012 and SA7038
10) NACA 0006 and S1223
11) NACA 0009 and S1223
12) NACA 0012 and S1223

Selected Airfoil Combination

NACA 0006 and NACA 4412

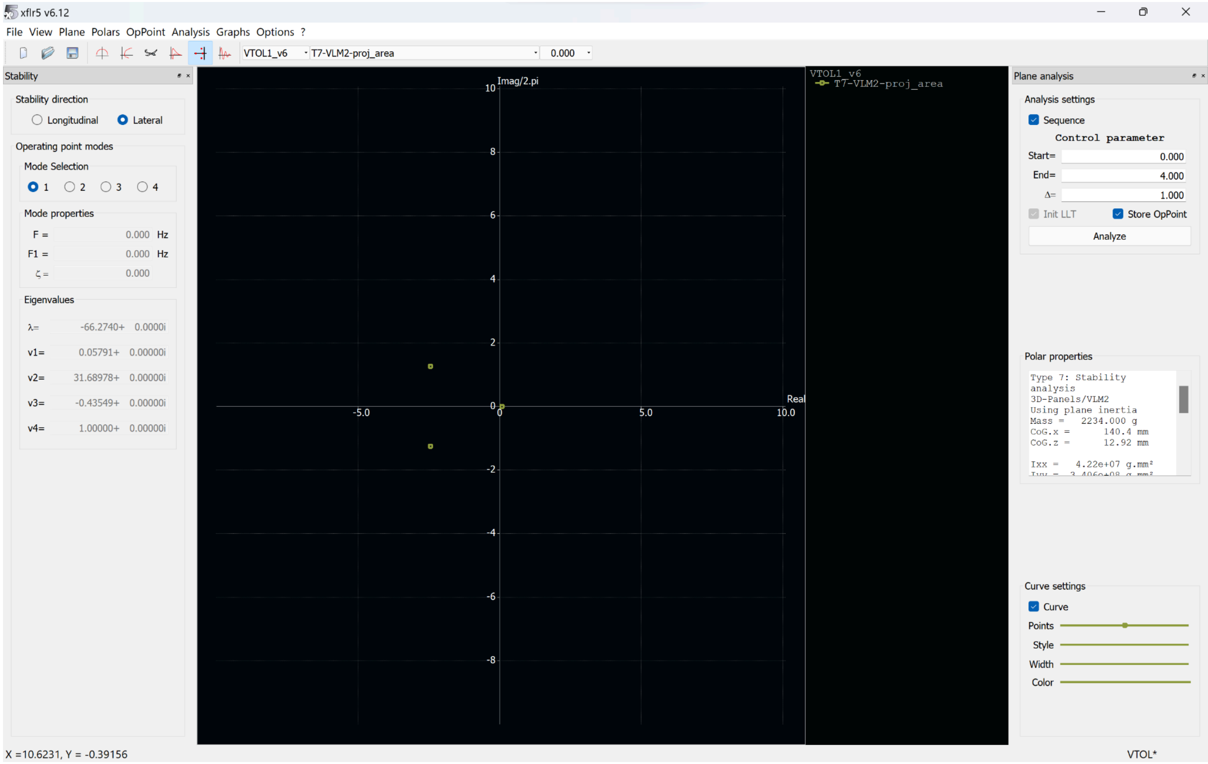
This airfoil combination was the most suitable in terms of various parameters such as L/D ratio, balanced AOA, design Cl, and cruise velocity. The values of these parameters for this airfoil combination are mentioned below:-

Design Cl = 0.295
Cl max = 1.85
Cruise velocity = 17.3 m/s
Balanced AOA = -0.6
L/D ratio = 13.879

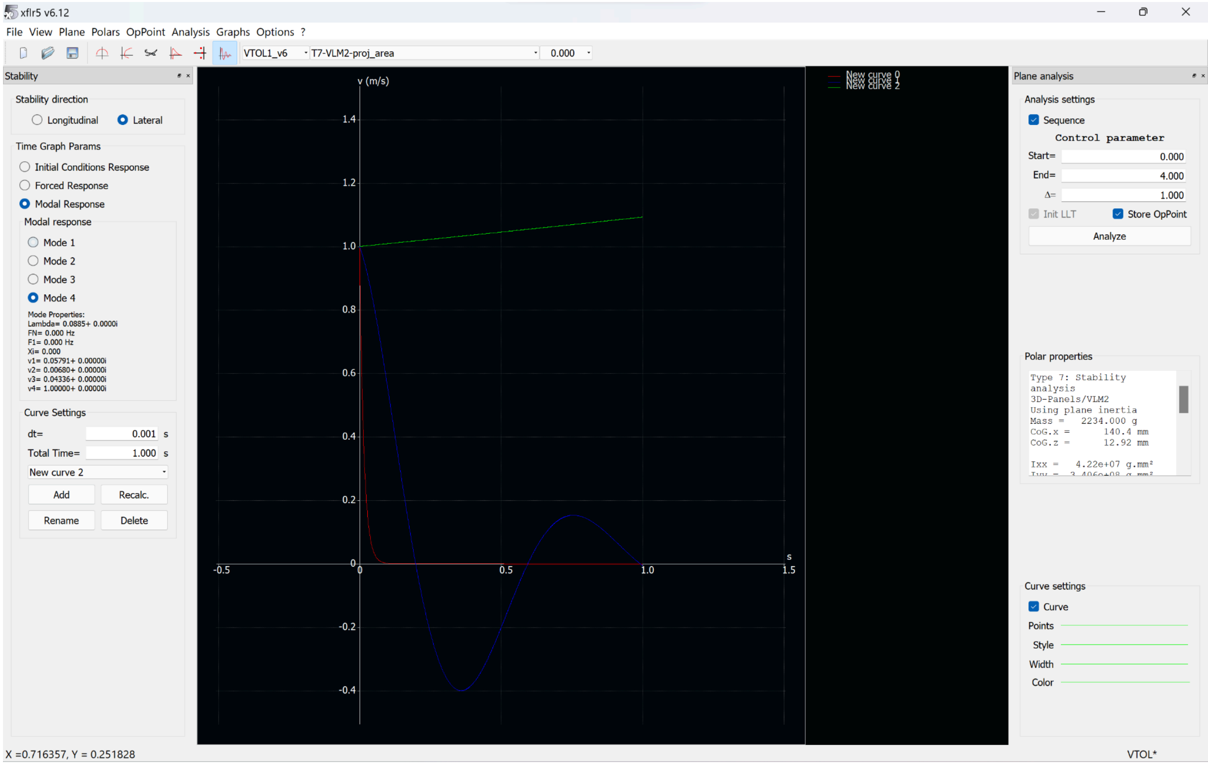
Overview of Stability Analysis:-

There are 2 Longitudinal modes: Short Period and Phugoid and 3 Lateral modes: Roll Damping, Dutch Roll, and Spiral

1) After verifying the stability prerequisites, we ran the stability analysis, first with zero gain. One point should appear in the polar graphs which would be the AOA at Cm=0 (which we checked from the log file). This point should lie on the Type-2 analysis done earlier and we were able to successfully observe it. 3 ways of observing the stability are 3D view, root locus graph, and time response analysis.
2) Root locus graph: More negative values mean more damping (as the real axis corresponds to the damping factor) and the imaginary axis corresponds to the frequency of oscillations. Positive values would mean undamped and unstable. Therefore, the presence of values more to the left corresponds to more stability. Modes lying on the real axis are non-oscillatory. Airfoil doesn’t have much influence on stability but mass, inertia, and geometry do.



3) Time response analysis: We plotted the various flight parameters vs. time. Longitudinal parameters: fluctuations in horizontal (u) and vertical (w) speed, pitch rate (q) and pitch angle (theta).
   Lateral parameters: lateral velocity variation (v), roll rate (p), yaw rate (r) and bank angle (phi).
   We performed modal response analysis for various modes.
   We also investigated the initial conditions and forced response analysis. We were able to observe the plots displaying the typical behaviors for their respective modes.



Stability mode notation of XFLR5:

Longitudinal:-
Mode 1: Short Period
Mode 2: Short Period
Mode 3: Phugoid
Mode 4: Phugoid

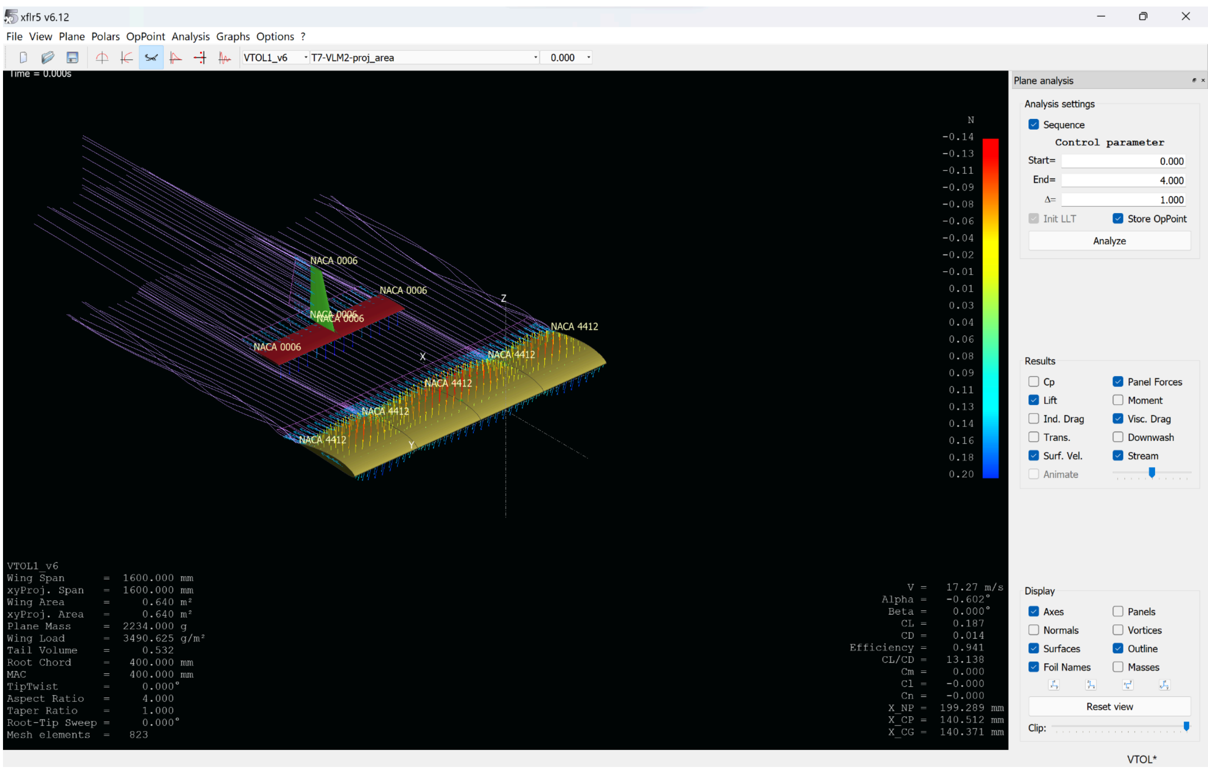
Lateral:-
Mode 1: Roll Damping
Mode 2: Dutch Roll
Mode 3: Dutch Roll
Mode 4: Spiral

Ideal characteristics of graphs for various modes:-

Short Period: high frequency, well-damped

Phugoid: slow, lightly damped, stable, or unstable
Roll Damping: high frequency, well-damped
Dutch Roll: faster and more damped than Phugoid
Spiral: non-oscillatory, slow, generally unstable

4) Sensitivity analysis: We analyzed the influence of inertia on stability by running a stability analysis, but this time by adding gain to the required inertia parameter. We ran a sequence analysis, checked the log file, and compared the eigenvalues (which correspond to points on the root locus graph) to understand the influence of that inertia parameter on the damping of the various modes. We also analyzed the polar plots as well as the root locus graph.
5) Control Derivatives: We observed the influence of various control parameters such as flap deflection on the stability. We verified which motions were being affected by that parameter and which were not.



The idea behind performing stability analysis is to find optimal values for various flight parameters such as tail distance, control surface dimensions, etc. for which the various flight modes are stable and we obtain minimum time response characteristics. We have primarily analyzed the variations of the various aerodynamic and flight parameters in the context of dutch roll. The reasons for not including the other modes in our analysis are as follows:-

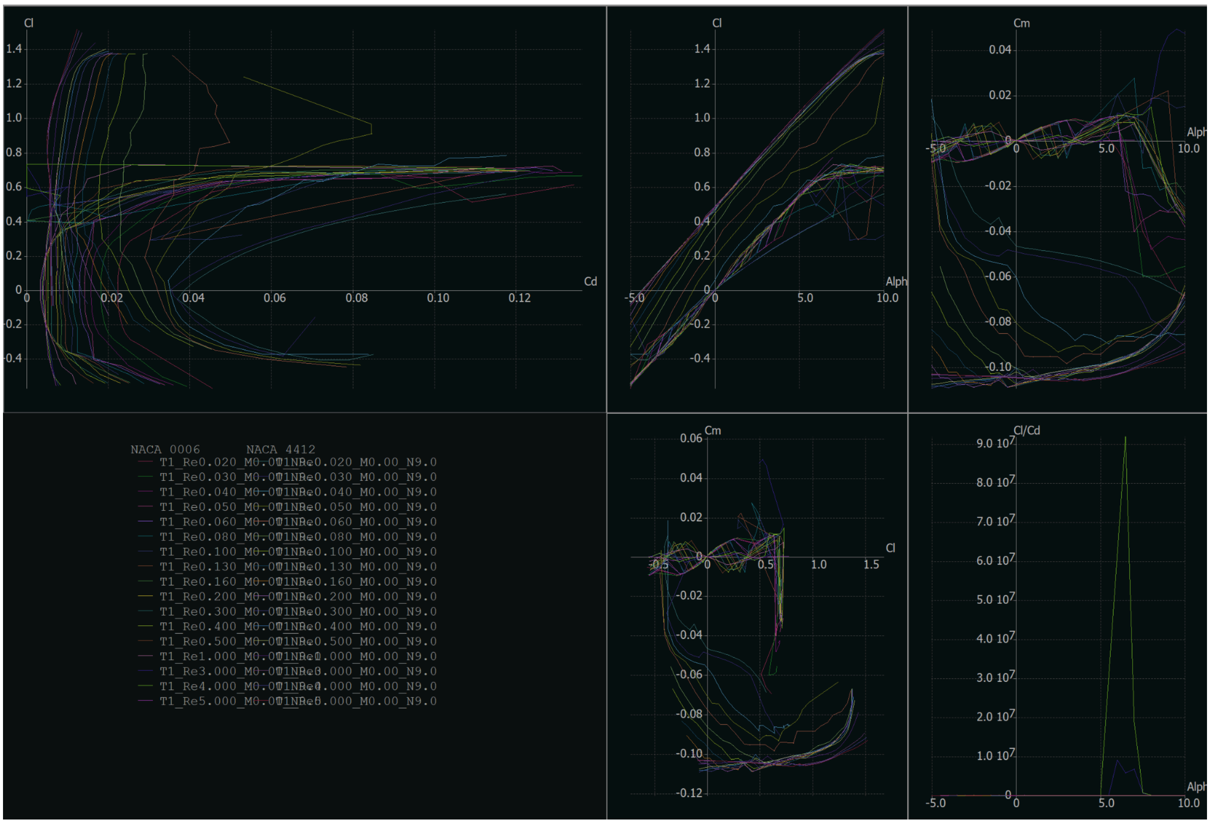
- Phugoid: This longitudinal stability mode in general is almost undamped/ slightly damped and our design has already achieved a stable configuration in this regard.
- Short Period: This is a highly damped mode with a very short settling time and therefore the minuscule variations observed in the short period dynamics are not of analytical significance.
- Roll Damping: As is the case with the short period mode, the highly damped nature of the roll damping mode renders it unfeasible to perform comparative analysis for variations in flight parameters.
- Spiral: The spiral dynamics that we observed were compliant with the conventional flight dynamics norms. The spiral response was non-oscillatory and divergent. This introduces a tendency in the aircraft to sideslip away from its heading. However, a simple fix to this and other similar problems that involve introducing sideslipping in the aircraft such as any unanticipated pressure gradient developing on the

rudder due to the flow stream emerging from the fuselage is to add a trim to the necessary control surfaces.

Dutch roll mode can be particularly sensitive to changes in parameters like tail distance, due to its inherent coupling between yaw and roll dynamics. Therefore, it provides clearer indications of how alterations in design features affect lateral stability compared to other modes.

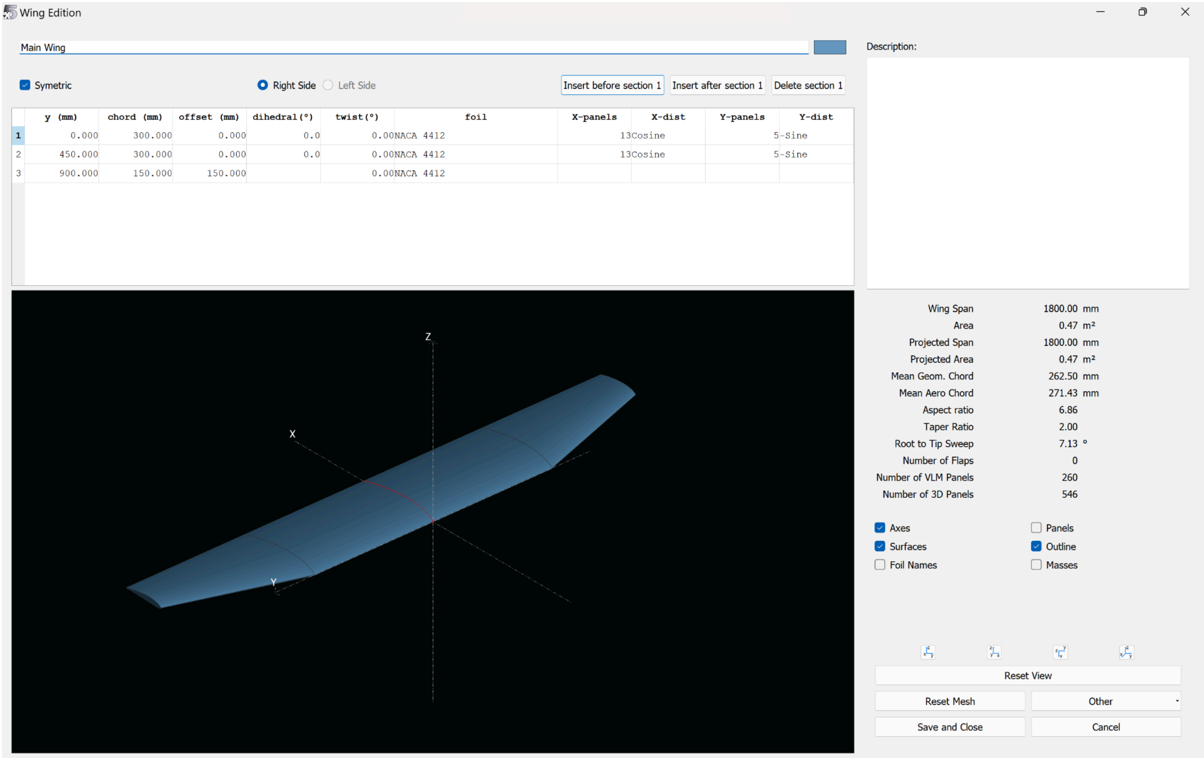
XFoil Direct Analysis:-

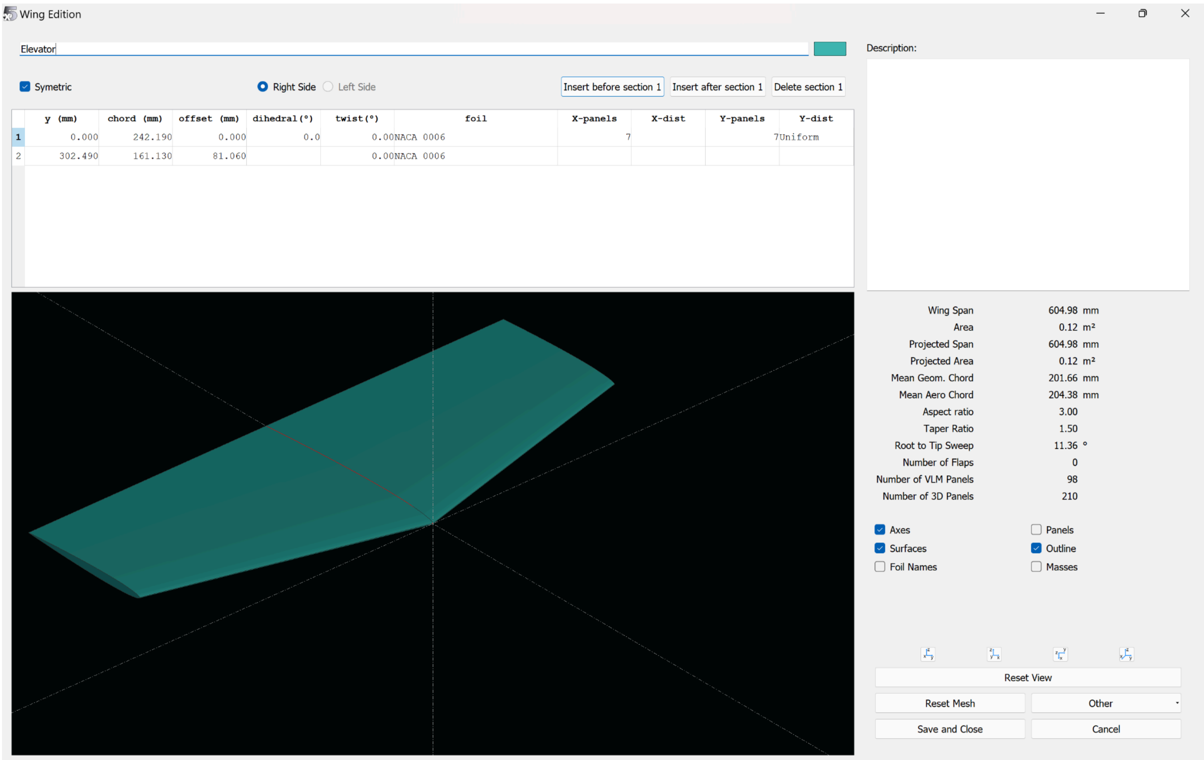
After finalizing the airfoil combination, we proceed with the direct analysis of these airfoils. Plots of various aerodynamic plots such as coefficient of lift (Cl), coefficient of drag (Cd), angle of attack (alpha), coefficient of pitching moment (Cm), and glide ratio (Cl/Cd) have been generated for our selected airfoils: NACA 4412 and NACA 0006 using multi-threaded batch analysis for Reynold’s number range from 20,000 to 5,000,000 and angle of attack range from -5 to 10 degrees.

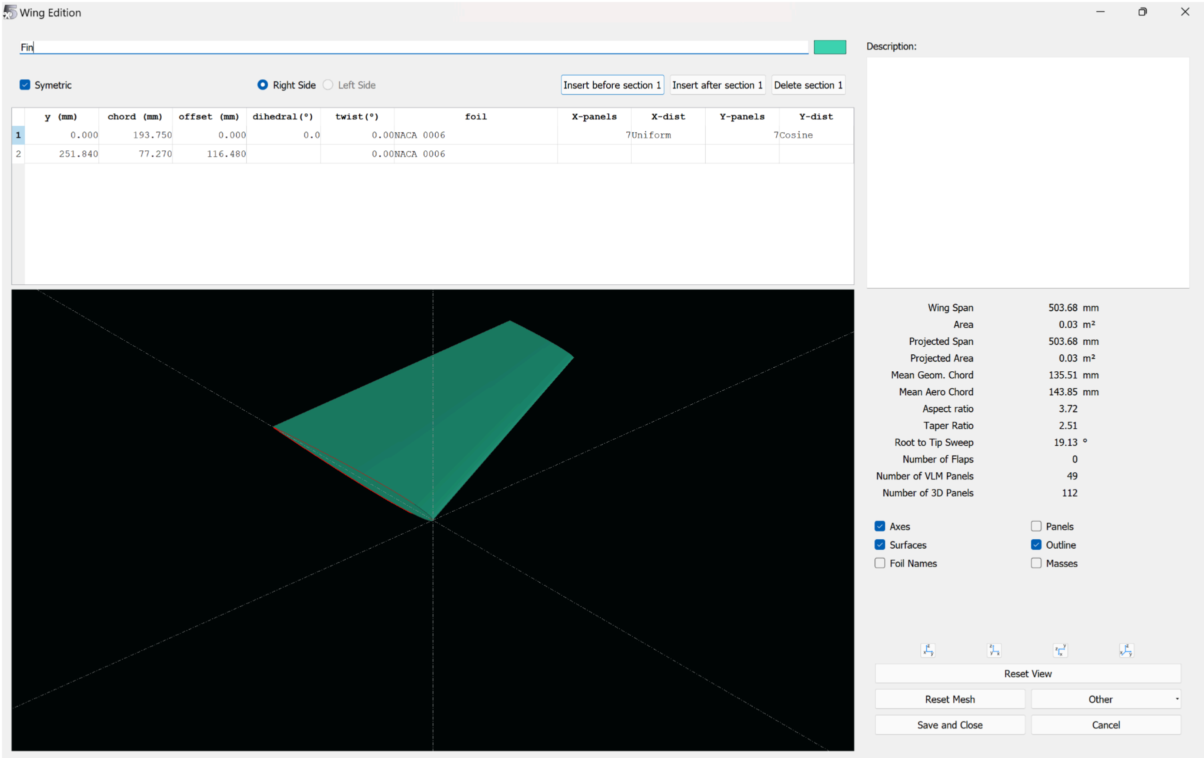


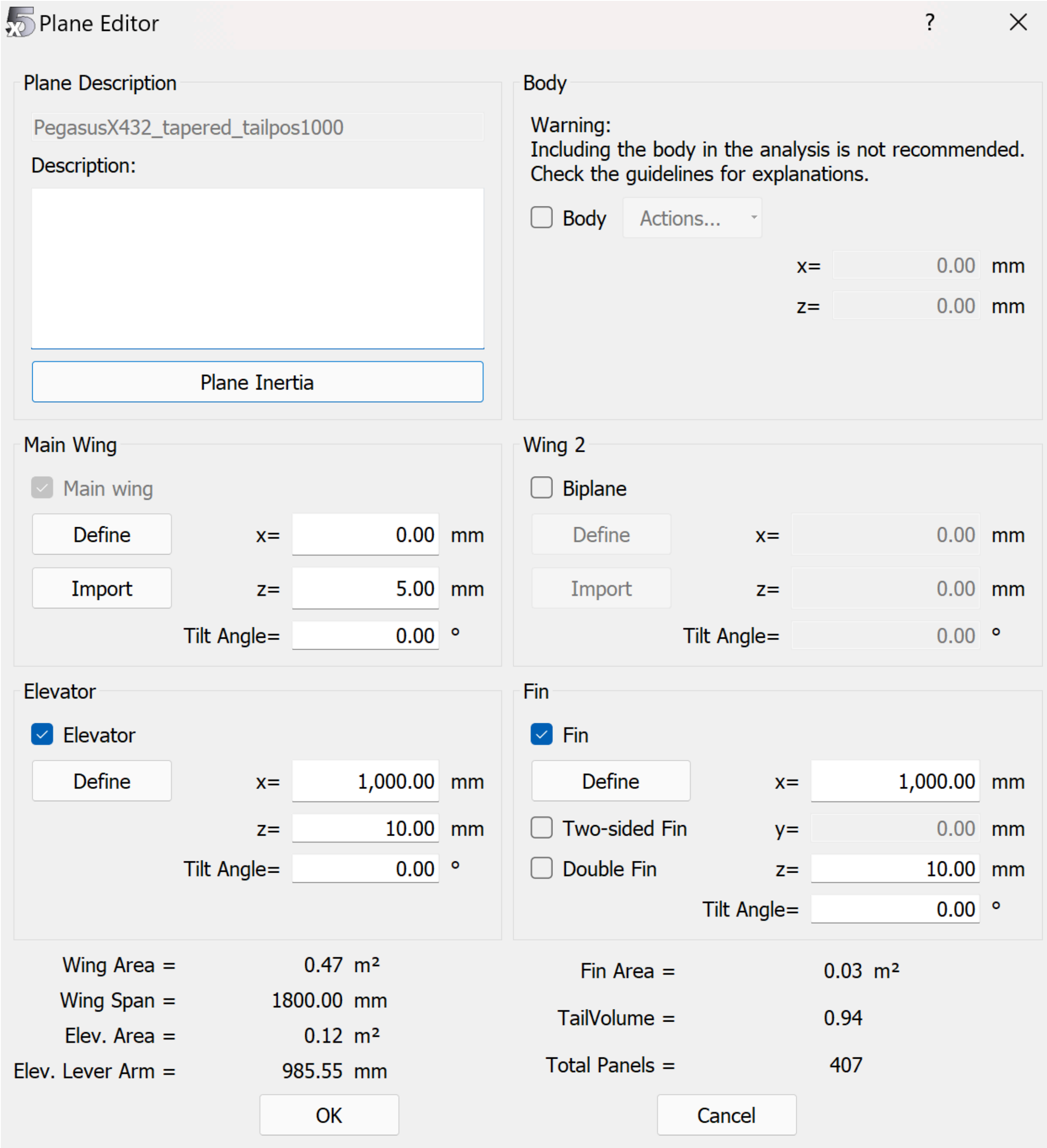
Wing and Plane design:-

We have used NACA 4412 for the main wing and NACA 0006 for the elevator and fin.


Wing Edition
Main Wing
Description:
Symetric
Right Side
Left Side
Insert before section 1
Insert after section 1
Delete section 1
y (mm) chord (mm) offset (mm) dihedral(°) twist(°) foil X-panels X-dist Y-panels Y-dist
1 0.000 300.000 0.000 0.0 0.00 NACA 4412 13 Cosine 5 -Sine
2 450.000 300.000 0.000 0.0 0.00 NACA 4412 13 Cosine 5 -Sine
3 900.000 150.000 150.000 0.00 NACA 4412
Z
X
Y
Wing Span 1800.00 mm
Area 0.47 m²
Projected Span 1800.00 mm
Projected Area 0.47 m²
Mean Geom. Chord 262.50 mm
Mean Aero Chord 271.43 mm
Aspect ratio 6.86
Taper Ratio 2.00
Root to Tip Sweep 7.13 °
Number of Flaps 0
Number of VLM Panels 260
Number of 3D Panels 546
Axes
Panels
Surfaces
Outline
Foil Names
Masses
Reset View
Reset Mesh
Other
Save and Close
Cancel

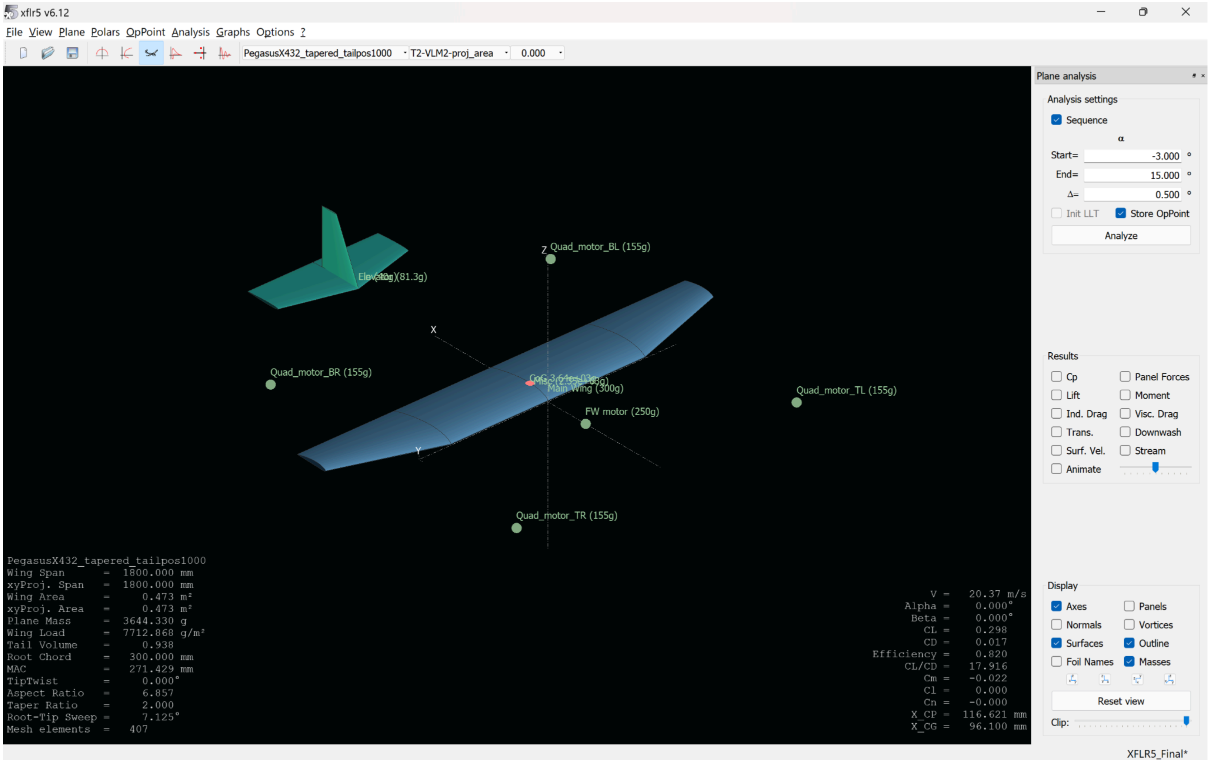

Wing Edition
Elevator
Description:
Symetric
Right Side
Left Side
Insert before section 1
Insert after section 1
Delete section 1
y (mm) chord (mm) offset (mm) dihedral(°) twist(°) foil X-panels X-dist Y-panels Y-dist
1 0.000 242.190 0.000 0.0 0.00 NACA 0006 7 7 Uniform
2 302.490 161.130 81.060 0.00 NACA 0006
Wing Span 604.98 mm
Area 0.12 m²
Projected Span 604.98 mm
Projected Area 0.12 m²
Mean Geom. Chord 201.66 mm
Mean Aero Chord 204.38 mm
Aspect ratio 3.00
Taper Ratio 1.50
Root to Tip Sweep 11.36 °
Number of Flaps 0
Number of VLM Panels 98
Number of 3D Panels 210
Axes
Panels
Surfaces
Outline
Foil Names
Masses
Reset View
Reset Mesh
Other
Save and Close
Cancel


Wing Edition
Fin
Description:
Symetric
Right Side
Left Side
Insert before section 1
Insert after section 1
Delete section 1
y (mm) chord (mm) offset (mm) dihedral(°) twist(°) foil X-panels X-dist Y-panels Y-dist
1 0.000 193.750 0.000 0.0 0.00 NACA 0006 7 Uniform 7 Cosine
2 251.840 77.270 116.480 0.00 NACA 0006
Wing Span 503.68 mm
Area 0.03 m²
Projected Span 503.68 mm
Projected Area 0.03 m²
Mean Geom. Chord 135.51 mm
Mean Aero Chord 143.85 mm
Aspect ratio 3.72
Taper Ratio 2.51
Root to Tip Sweep 19.13 °
Number of Flaps 0
Number of VLM Panels 49
Number of 3D Panels 112
Axes
Panels
Surfaces
Outline
Foil Names
Masses
Reset View
Reset Mesh
Other
Save and Close
Cancel



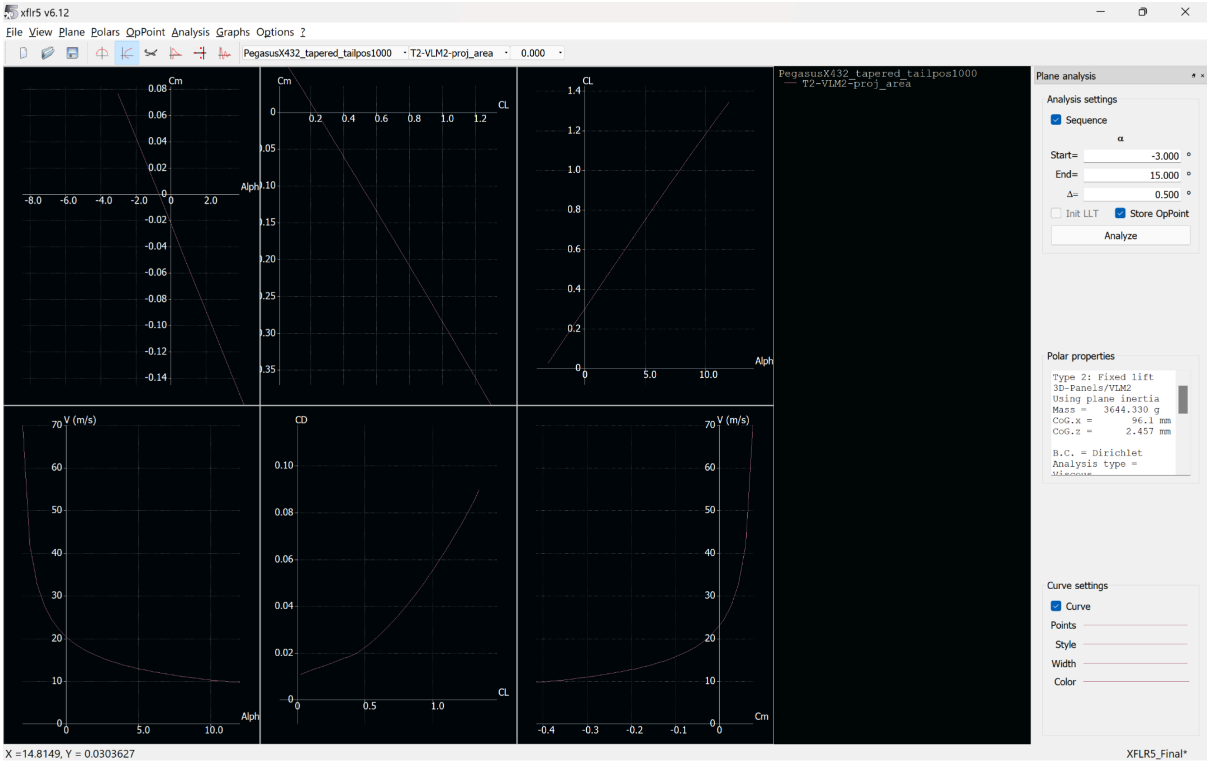
We have given an almost negligible z elevation of 5 mm to the main wing and 10 mm to the elevator and fin just to prevent the Goldstein singularity error (division by zero error). We have given different elevations to the main wing and the elevator and fin so that the airflow emerging from the main wing is not perturbed by the elevator and fin in the computational model and hence we will be able to accurately calculate the flow characteristics.

Mass distribution:-



Mass (g)	x (mm)	y (mm)	z (mm)	Description
155.000	-575.000	-650.000	0.000	Quad_motor_TL
155.000	-575.000	650.000	0.000	Quad_motor_TR
155.000	725.000	-650.000	0.000	Quad_motor_BL
155.000	725.000	650.000	0.000	Quad_motor_BR
250.000	-200.000	0.000	0.000	Fixed Wing motor
2,353.000	75.000	0.000	0.000	Miscellaneous

We have performed a Type-2 (fixed lift) analysis on this wing and plane setup using the ring vortex (VLM2) method and wing planform projected on the xy plane. We have run a sequential analysis from the angle of attack -3 to 15 degrees with increments of 0.5 degrees.



We have obtained these aerodynamic plots of the various polar properties.

The stability prerequisites have been verified as the Cm vs alpha slope is negative indicating that the plane is positively stable and the lift at zero Cm is also positive.

We calculated the neutral point from the Cm vs alpha curve as it is the point where this curve becomes horizontal.

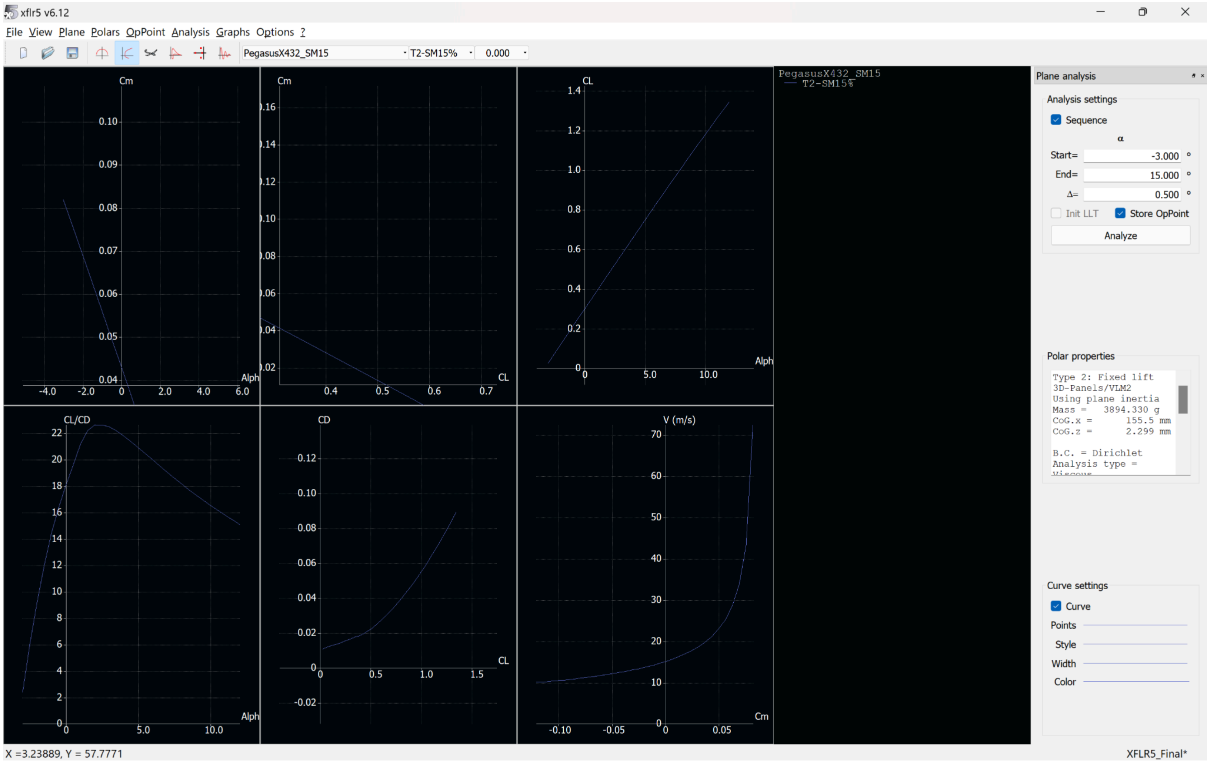
Using hit and trial, we were able to run the sequential analysis enough times to obtain the neutral point.

We have identified the neutral point to be 196 mm from the leading edge of the main wing. Considering a 15 % static margin, we obtain the position of CG as 155 mm from the leading edge.

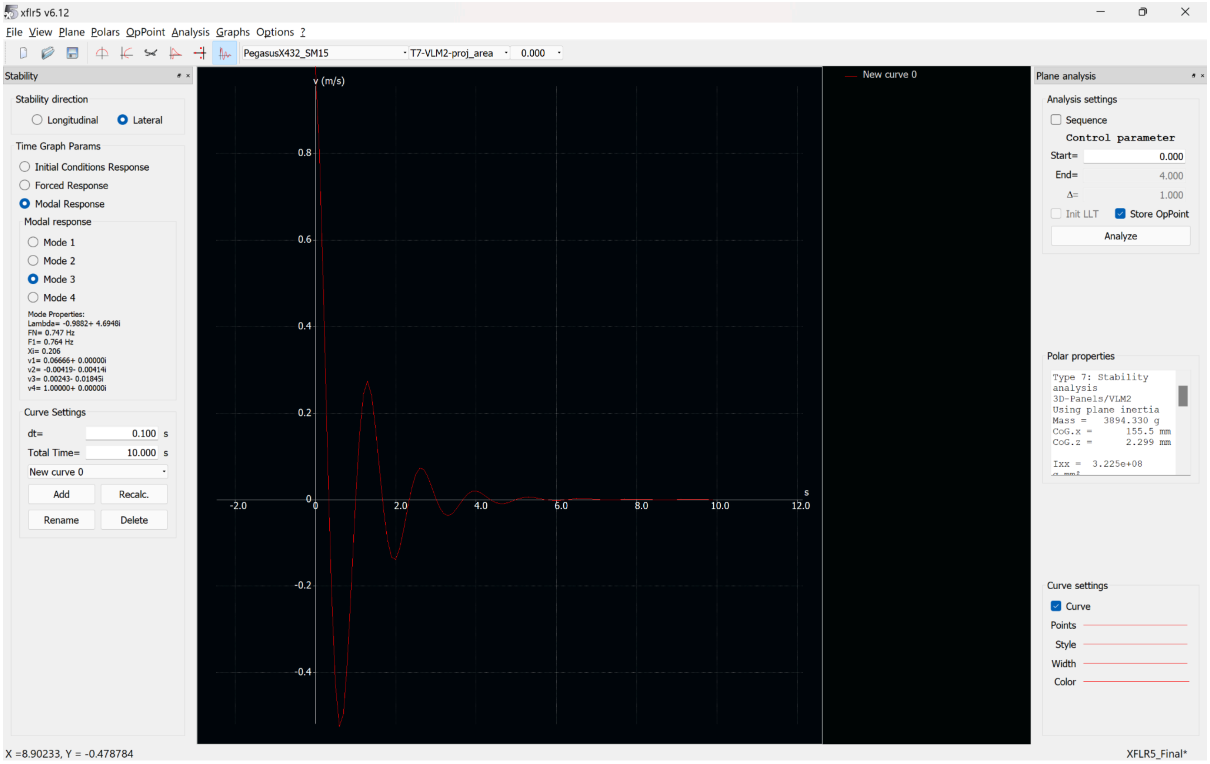
New mass distribution:-

Mass (g)	x (mm)	y (mm)	z (mm)	Description
155.000	-495.000	-650.000	0.000	Quad_motor_TL
155.000	-495.000	650.000	0.000	Quad_motor_TR
155.000	805.000	650.000	0.000	Quad_motor_BR
155.000	805.000	-650.000	0.000	Quad_motor_BL
250.000	-200.000	0.000	0.000	FW motor
2,353.000	146.000	0.000	0.000	Miscellaneous
250.000	155.000	0.000	0.000	Payload

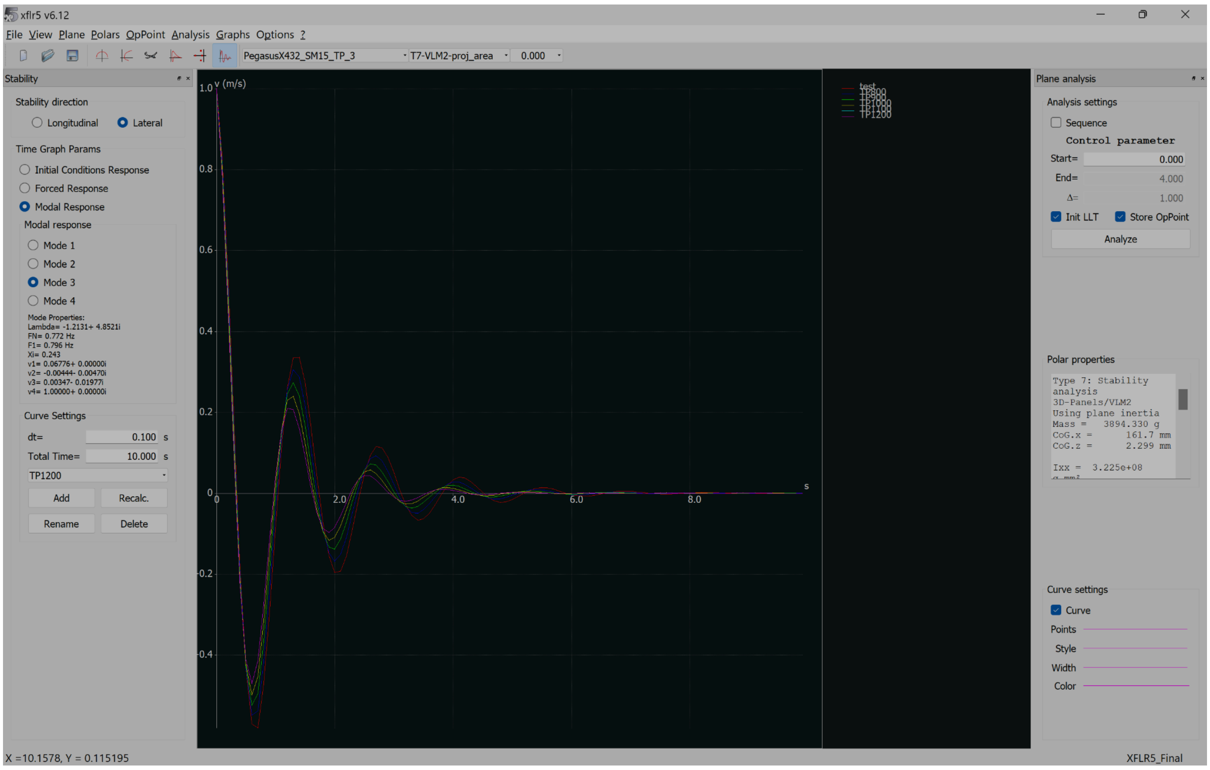
On performing the type 2 sequential analysis for this 15% static margin setup, we obtained the following plots:
In conclusion, the final parameters that we have obtained from our XFLR5 analysis are:
Balanced angle of attack = 3.2 degrees
Design Cl = 0.3
Cdmin = 0.01
Cruise velocity = 15 m/s
L/D ratio = 15.069



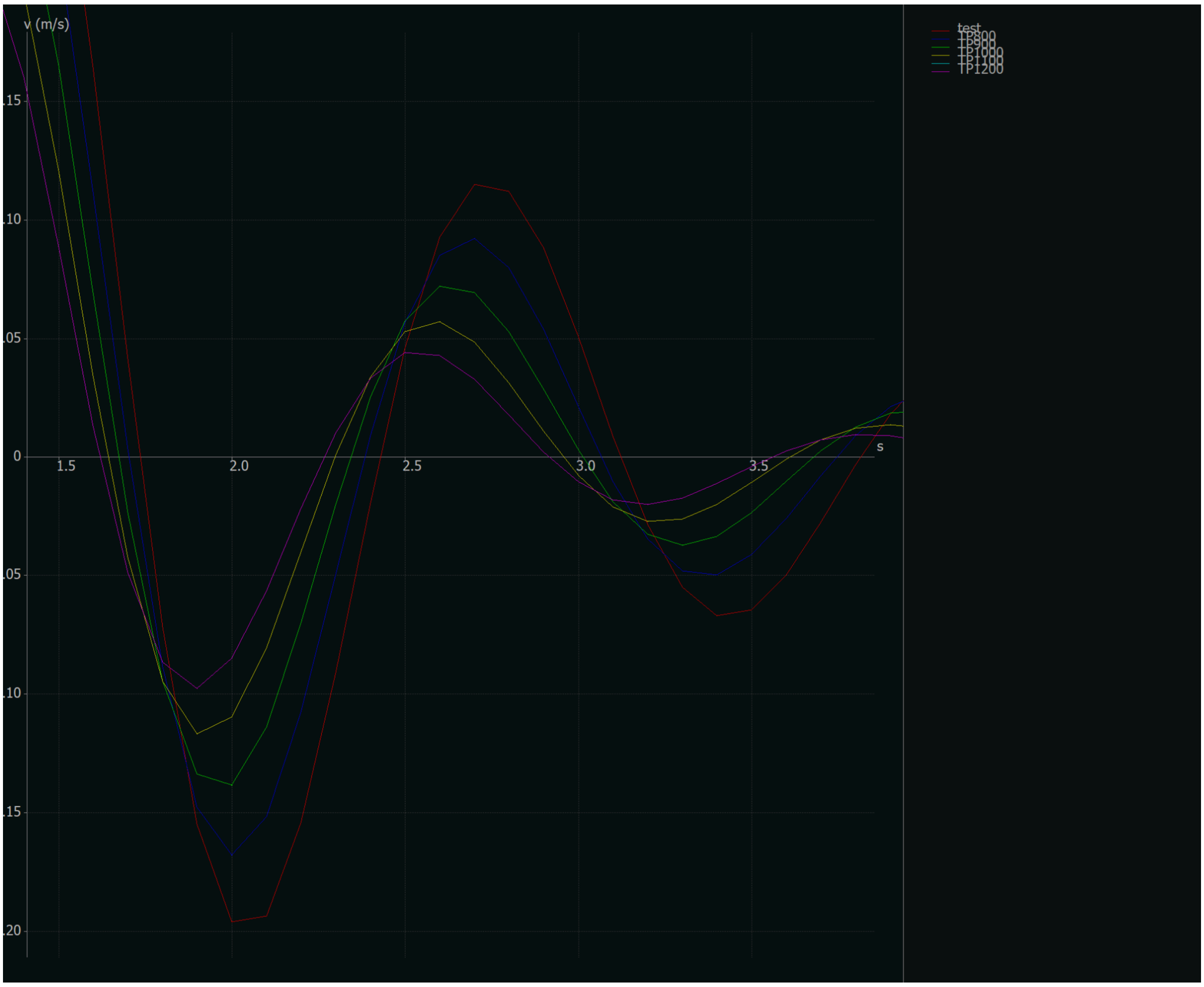
Our stability prerequisites remain satisfied. On performing type 7 stability analysis, we got the expected root locus plots and time response graphs for the longitudinal and lateral stability modes. We obtained a very small dutch roll time of 6 seconds which indicates strong lateral stability and the self-stabilizing nature of the plane.



We have compared the dutch roll time response analysis for different tail positions (800, 900, 1000, 1100, and 1200 mm from the main wing). Here are the obtained results:-

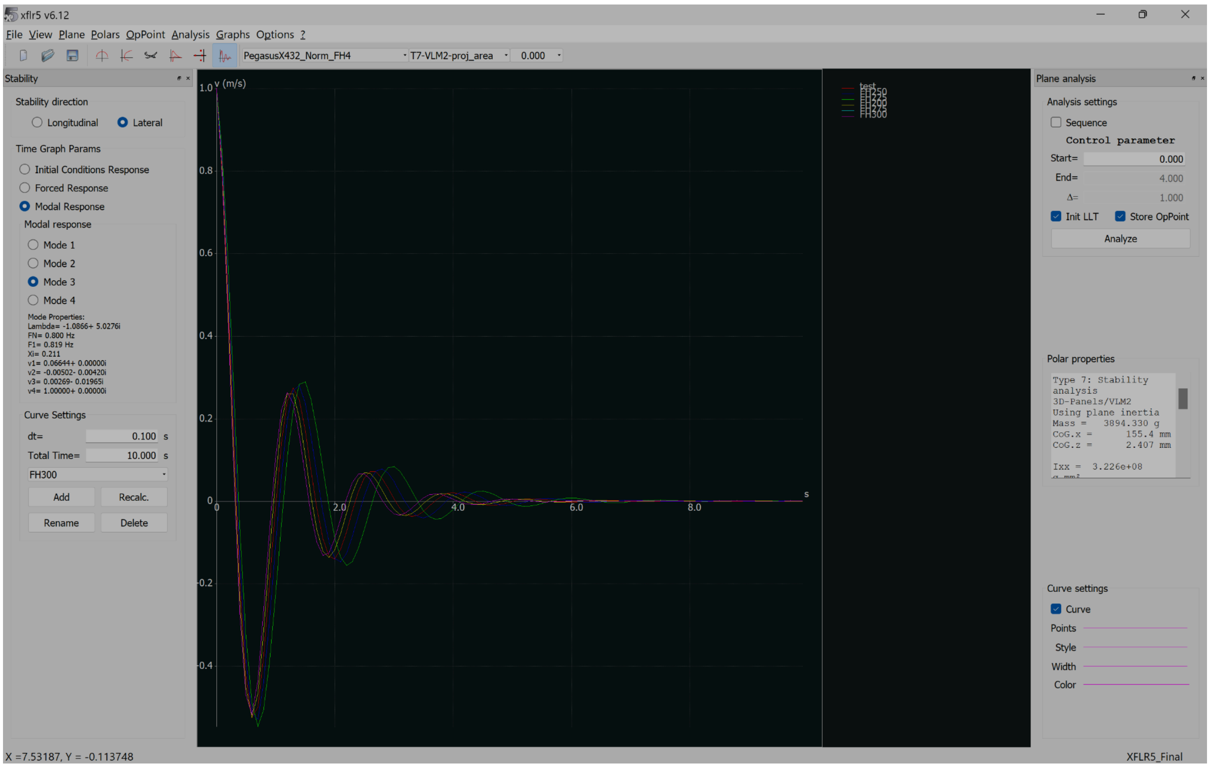


Legend:-
TP800 - red
TP900 - blue
TP1000 - green
TP1100 - yellow
TP1200 - pink



From the close-up of the comparison plots, it is clear that as the tail position increases, the dutch roll time response curve dampens faster. However, after a certain point, by increasing the tail position, we are inadvertently increasing the aircraft size. Moreover, as our design includes a carbon fiber rod connecting the fuselage and main wing to the tail, increasing the tail distance beyond proportion leads to huge bending moments acting on the connecting rod which will inevitably cause the rod to fail. To prevent this, we have chosen an optimal value of the tail position (1000 mm from the leading edge of the main wing) to keep a balance between the stability and the structural strength of our design.

We have compared the dutch roll time response analysis for different fin heights (200, 225, 250, 275, and 300 mm). Here are the obtained results:-



Legend:-

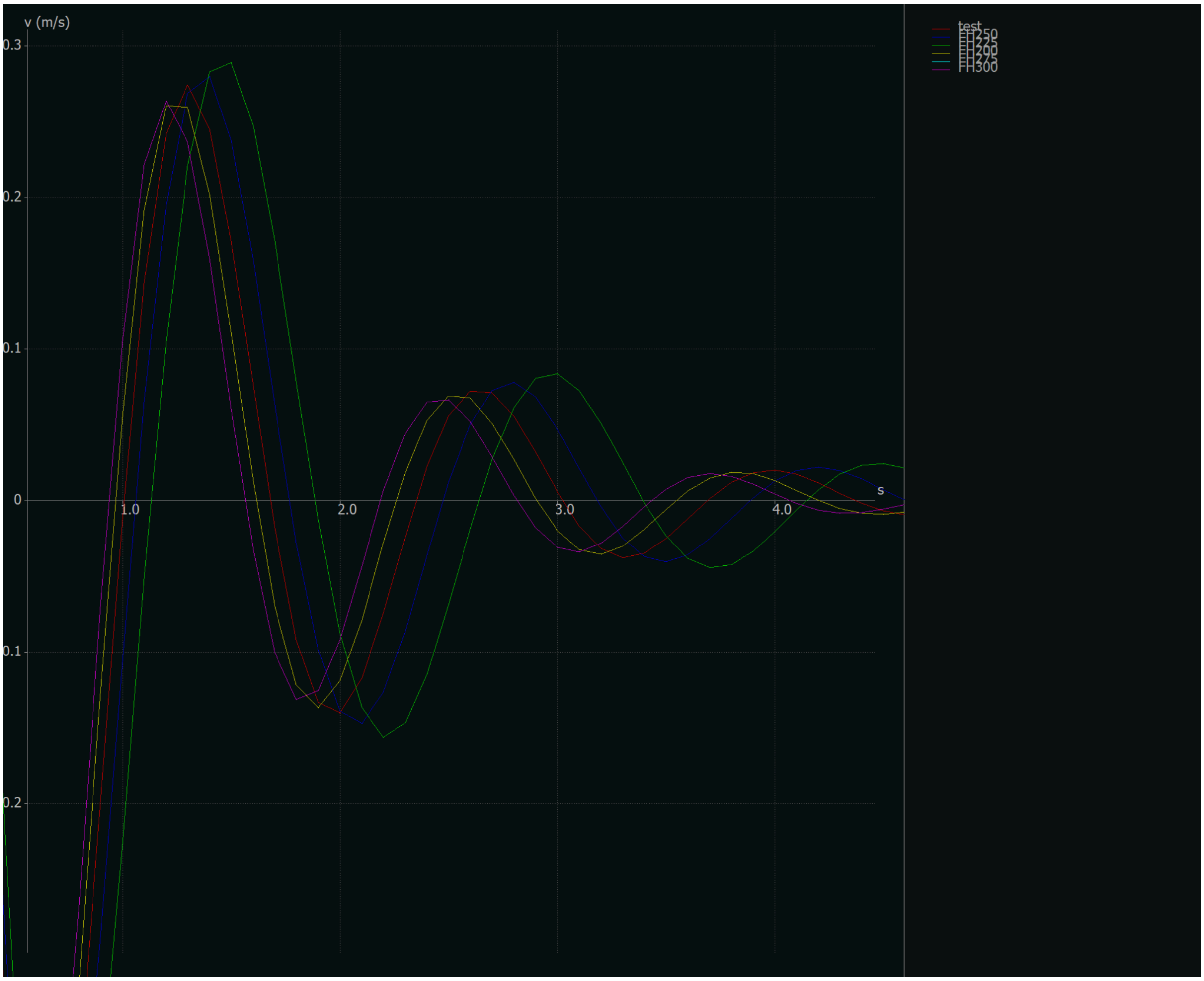
FH250 - red

FH225 - blue

FH200 - green

FH275 - yellow

FH300 - pink



From the close-up of the comparison plots, it is clear that as the fin height increases, the dutch roll time response curve dampens faster. As we keep increasing the fin height, the loading conditions on the fin increase greatly due to the bending moments acting on the fin. Therefore, as perpetually increasing the fin height leads to its eventual failure due to the intense structural loading, we have identified an optimal fin height that ensures good stability characteristics while also maintaining the structural integrity of our design.

We have compared the dutch roll time response analysis for different sweep angles of the fin (0, 5, 10, 15, 20, 25, 30, 35, and 40). Here are the obtained results:-

Legend:-

SA0 - red

SA5 - dark blue

SA10 - green

SA15 - yellow

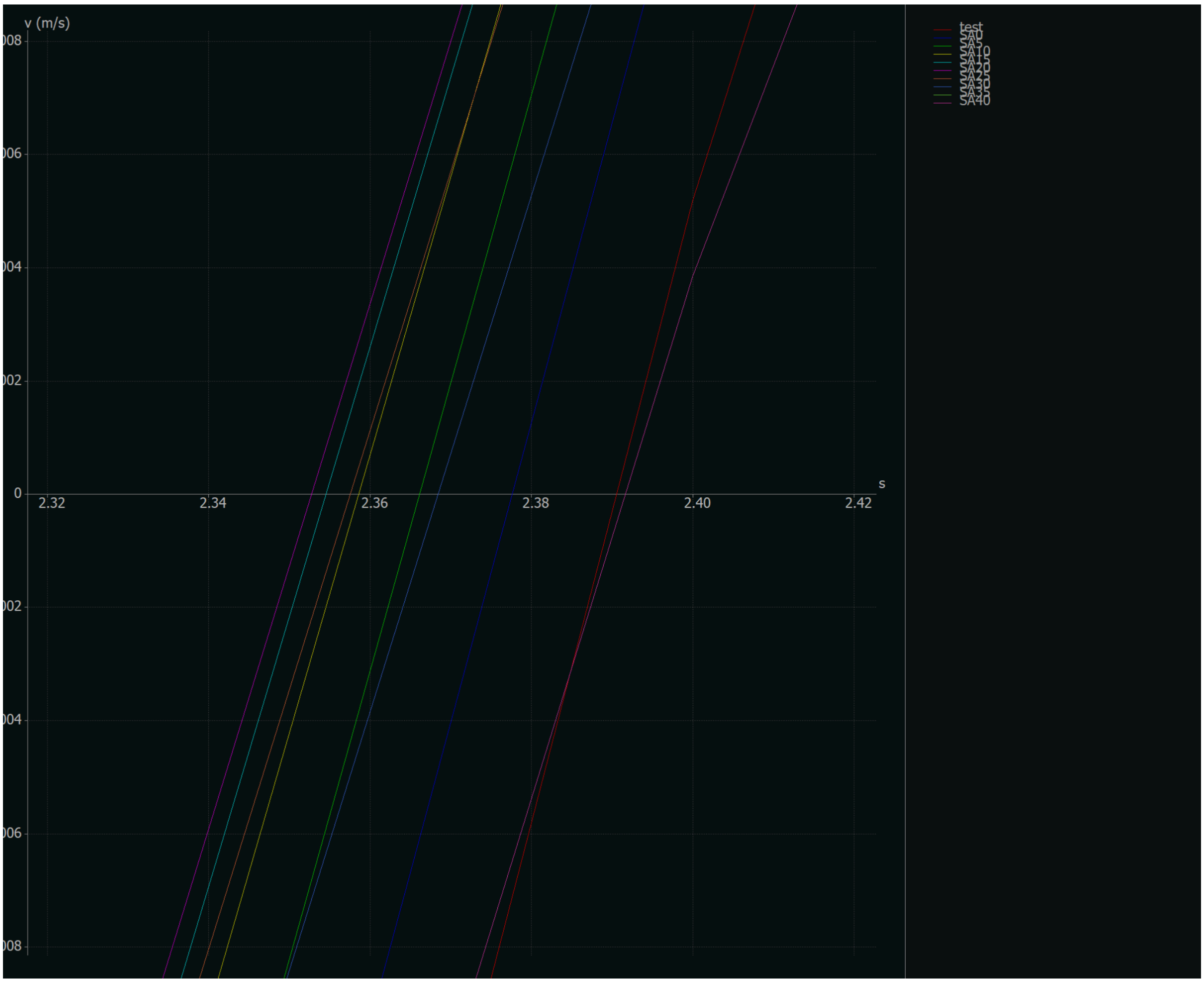
SA20 - light blue

SA25 - purple

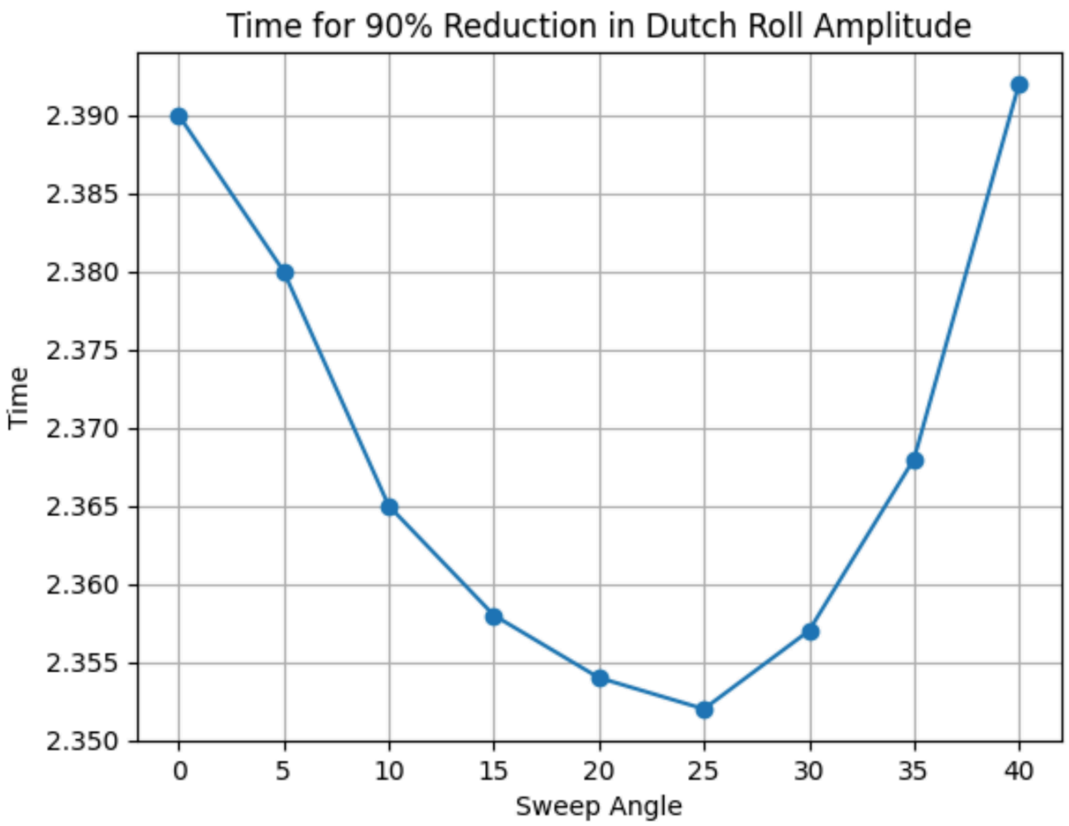
SA30 - brown

SA35 - medium blue

SA40 - pink



From the close-up of the comparison plots, we can observe that the dutch roll time response dampens faster as the sweep angle increases, but only till 25 degrees. Once the sweep angle exceeds 25 degrees, the time required for dutch roll damping starts to increase. Therefore, we have chosen a sweep angle of 25 degrees for the fin as this allows us to achieve minimum dutch roll damping time.


Time for 90% Reduction in Dutch Roll Amplitude
Time
2.390
2.385
2.380
2.375
2.370
2.365
2.360
2.355
2.350
0
5
10
15
20
25
30
35
40
Sweep Angle