# Introduction to XFLR5
AME40451 – Aerospace Dynamics
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XFLR5 is a computational tool for understanding the stability and performance of small, model-sized aircraft operating at low Reynolds numbers. The workflow consists of three main sections. The first is airfoil design, which involves creating or importing 2D airfoils that will be used for the aircraft. The second stage is aircraft design. In this stage, you will define your aircraft's details, such as wingspan, chord length, airfoil locations, and shapes. Third is the analysis stage; in this section, you will take the aircraft that you designed in the second stage and perform numerical tests to determine how it reacts to certain conditions, such as changes in flow speed or angle of attack. In this guide, you will work through the basics of the software and create your own sample aircraft to test in XFLR5.

Section 1: Downloading and Installing the Software

1. Go to: http://www.xflr5.tech/xflr5.htm and click on "Downloads."

2. You will be directed to https://sourceforge.net/projects/xflr5/files/

3. Please download the latest version of the software available

4. After downloading, uncompress the files

5. In the folder, there should be an application named "xflr5" with the software logo; this is the main application. Therefore, click this to run the software as shown below
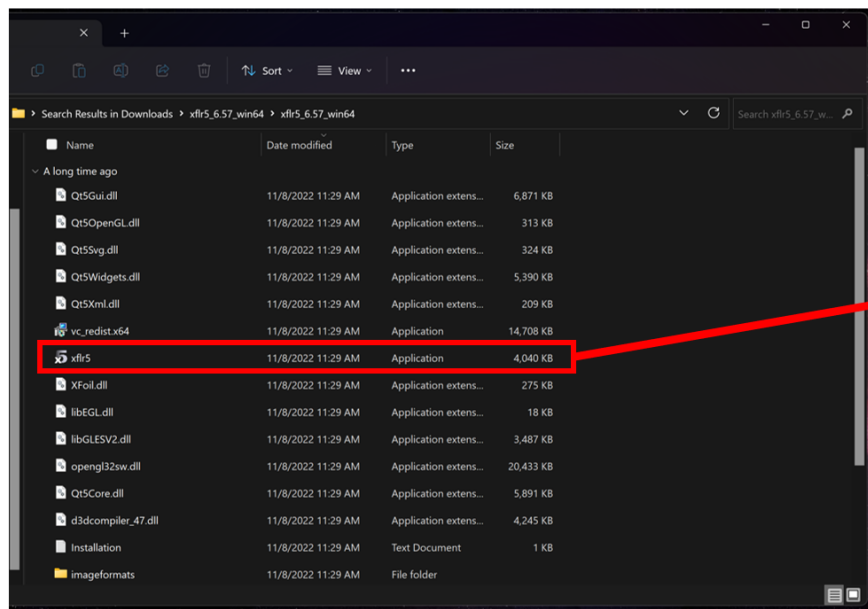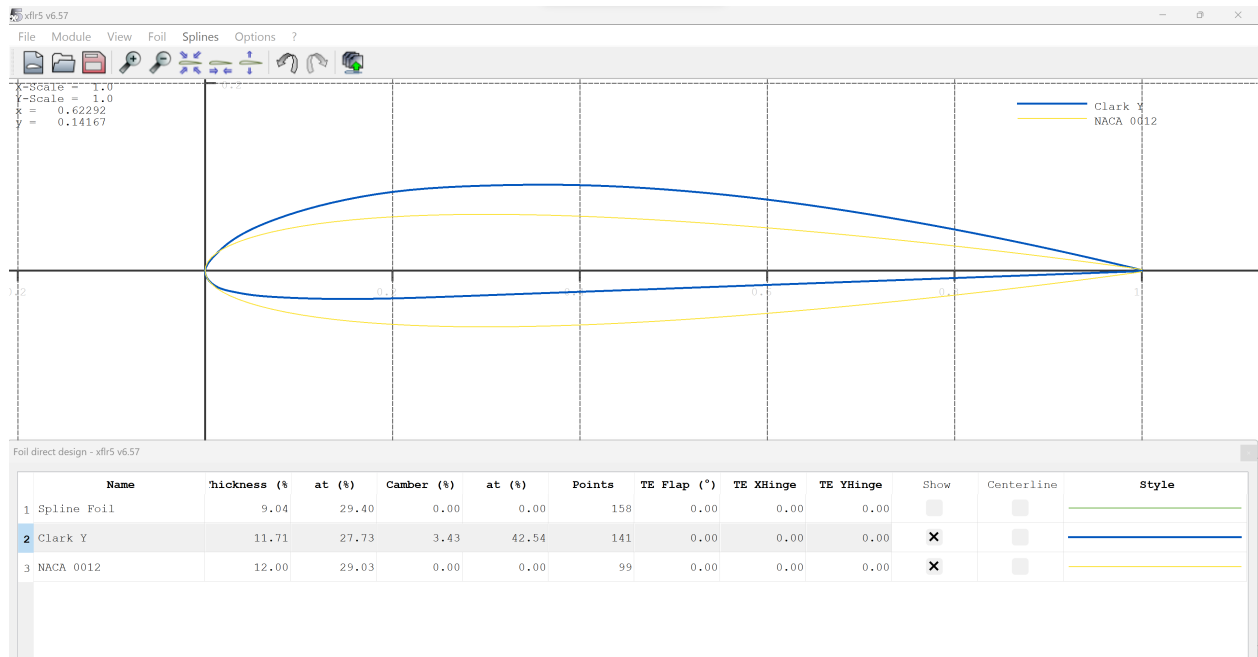
Section 2: Working with 2D Airfoils

1. After opening, you will be faced with a blank screen with only a toolbar at the top of the screen.
2. To begin, click "Module" > "Direct Foil Design."
3. To create a NACA airfoil, click "Foil" > "Naca Foils", and enter the 4-digit NACA code "0012" to create a symmetric airfoil. (Use default 100 panels.)
4. Click OK to create the airfoil. Click OK again to keep it named NACA 0012. You should see a nice symmetric airfoil on your screen.
5. Next, you will import a custom airfoil. Click the open button in the upper-left corner and select the "Clark_Y_Coordinates.dat" file you downloaded from Canvas. This file contains the coordinates that define the airfoil.
6. Locate the Clark Y airfoil in the list on the bottom of the screen, and rename it "Clark-Y". (Right-click it and select "Rename".)

Pro-tip: For future reference, common airfoil geometry files can be found online here:

http://airfoiltools.com/

7. Make sure that you have a NACA 0012 airfoil and Clark-Y airfoil on the list at the bottom of the screen.

	Name	Thickness (%	at (%)	Camber (%)	at (%)	Points	TE Flap (°)	TE XHinge	TE YHinge	Show	Centerline	Style
1	Spline Foil	9.04	29.40	0.00	0.00	158	0.00	0.00	0.00			
2	Clark Y	11.71	27.73	3.43	42.54	141	0.00	0.00	0.00	✕		
3	NACA 0012	12.00	29.03	0.00	0.00	99	0.00	0.00	0.00	✕		

8. The next step is to perform a batch analysis on the airfoils that have been selected. The batch analysis allows the system to record information about the flow around airfoils so that it can be used when analyzing the stability of the aircraft. To begin, click "Module" > "Xfoil Direct Analysis." In the new winow, select "Analysis" > "Batch Analysis."

9. On this screen, the system will determine the Reynolds numbers to test the airfoil. For this, you will use the default Reynolds numbers. Shift-click to high-light both airfoils and adjust the range of angle of attack values: Alpha Min = -15, Max = 15, and Increment -0.5.

10. Make sure the option "Store operating points" is selected. Then click the "Analyze" button at the bottom to run the Xfoil numeric solver.
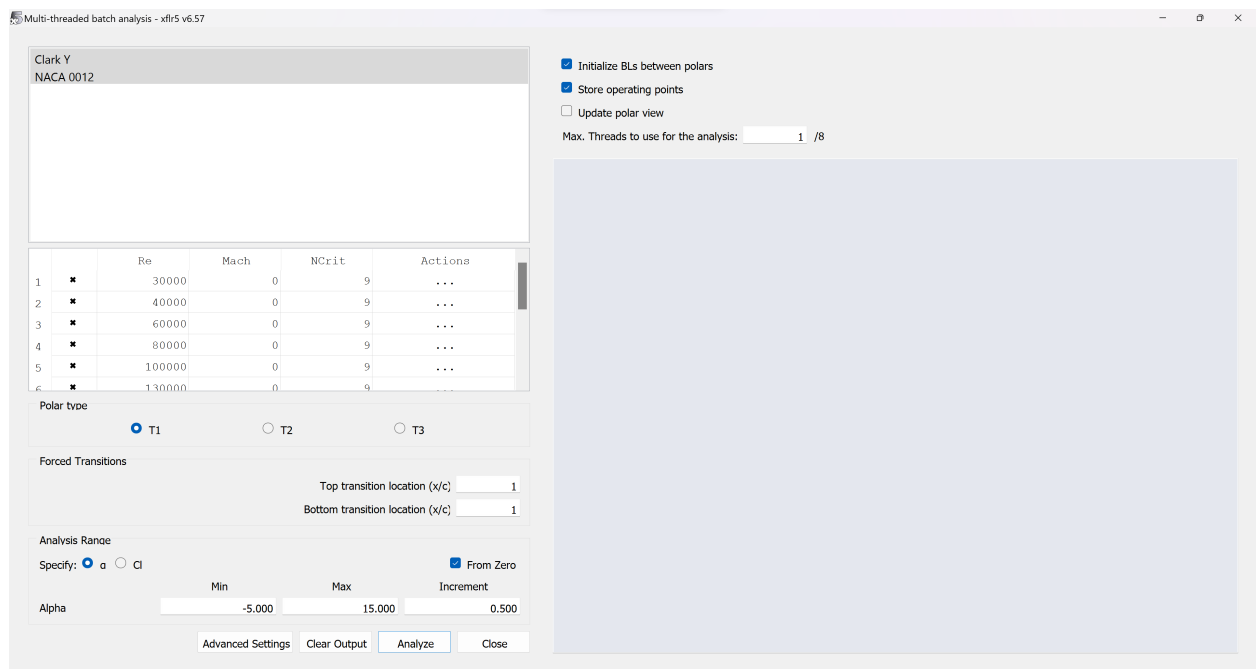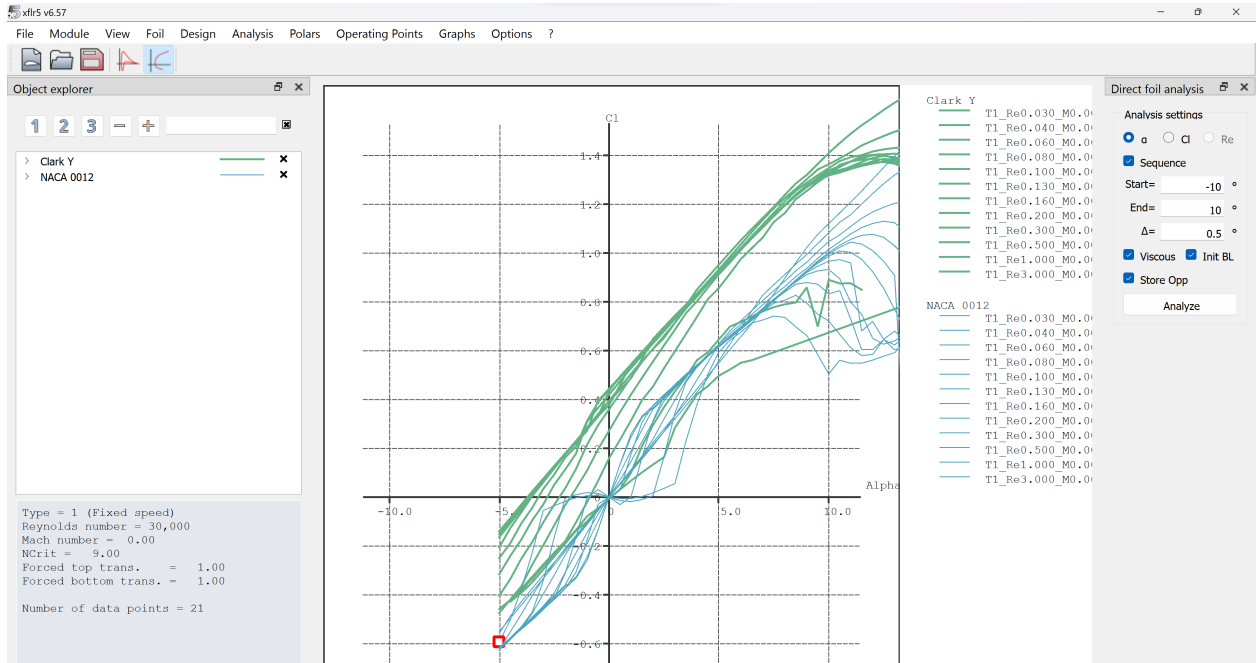
11. Close the window when the analysis is complete, and you should see plots similar to the ones pictured above. These are the results of the airfoil analysis for both the NACA 0012 and the Clark-Y. Please take time to change the graph results and review some of the information the system calculates.

   a. Right-click the largest graph and select "Current Graph" > "Define Graph Settings."

   b. Make the largest graph plot the lift force coefficient as a function of angle of attack.

   c. Note the color coding of the curves. Which curves correspond to the symmetric airfoil, and which correspond to the cambered Clark-Y? Consider the plot of Cl vs Alpha. Do the results make sense?

Deliverable 1: Take a single screenshot of all the Xfoil output graphs. Upload it to the appropriate location on the Canvas assignment.

12. Now that the 2D airfoil information is calculated, the next step is to design the aircraft for analysis.
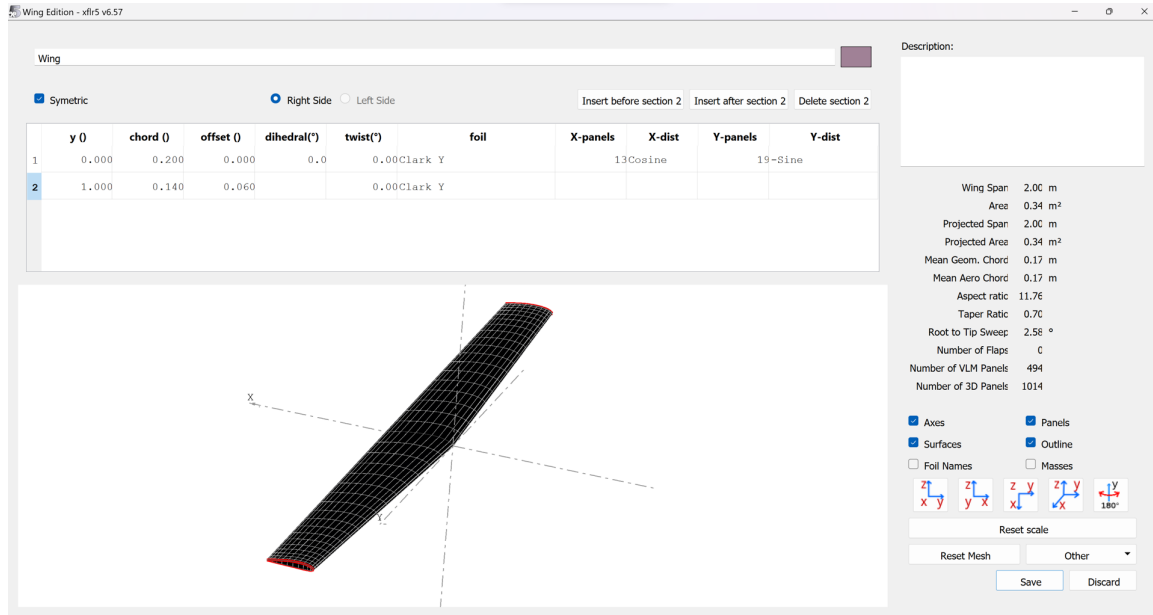
Section 3: Creating an Aircraft

1. To begin creating an aircraft, click "Module" > "Wing and Plane Design".

2. Next, click "Plane" > "Define a New Plane." A new winow called the "Plane Editor" should appear that looks like the one shown below. This is the main screen where you will define the dimensions of your plane.
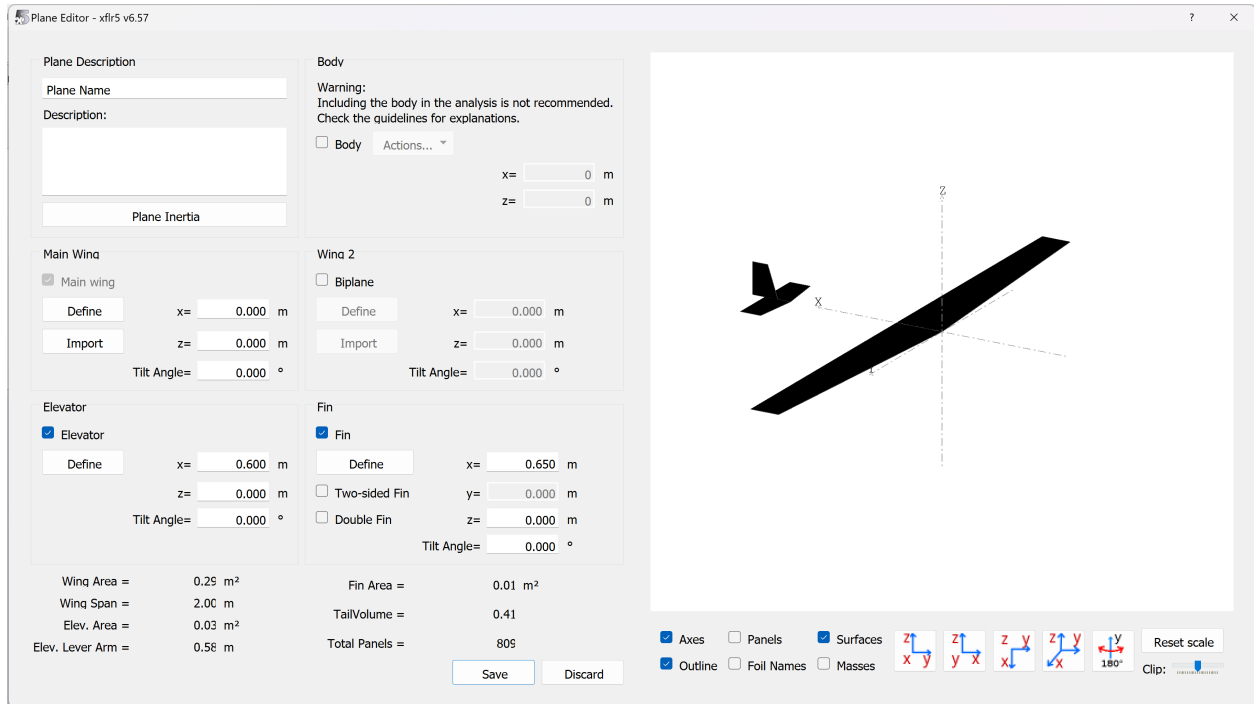
3. To define the dimensions of the main wing, click on the "Define" button below the "Main Wing" section to get a similar screen to the one shown below.

4. XFLR5 wing design operates by defining cross sections of the wing and then defining a mesh between those sections. To see how this works, click on the box in the "foil" column and select the "Clark-Y" airfoil that you imported previously.

   a. Do this for both boxes. What this is doing is defining the cross-section at the tip and root of the wing to both be a Clark-Y. You should see the wing turn into a 3D model after you do this.

   b. The top row defines the root geometry and the bottom row defines the wing tip geometry.
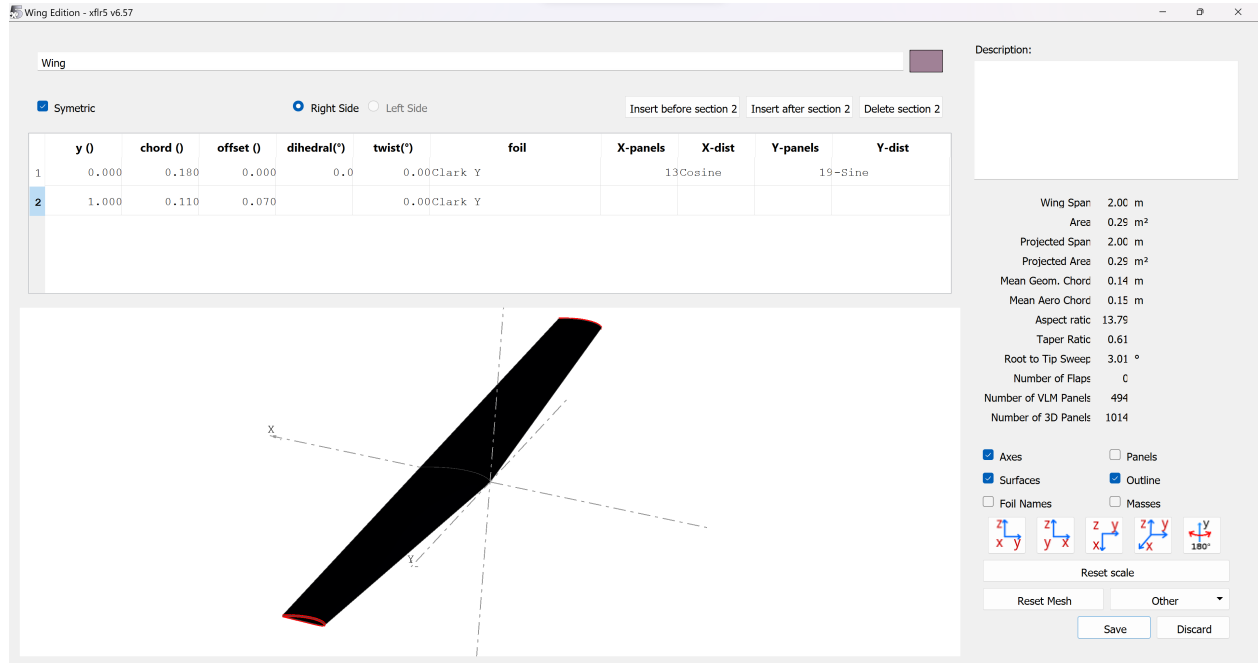
5. Next, define the span of the wing by setting the "y()" coordinates. Set the "y()" value in the second row to 0.8 m (800 mm) so that our model aircraft has a span of 1.6 m (1600 mm).

6. Set the chord length of the root cross-section (first row) to 0.2 m (200 mm).

7. To create a taper, set the tip cross-section chord (second row) to 0.14 m (140 mm). This will create a difference in chord lengths that XFLR5 will interpret as a taper and create a wing with a taper ratio of 0.7.

8. Notice that the trailing edge is not straight. This is because the offset of the cross-section is misaligned with the rest of the wing. To straighten the trailing edge, we must determine the new offset. In this case, the difference between chords is 0.2 - 0.14 m; therefore, our new offset is 0.06 m (60 mm).

9. Finally, check the "Panels" box on the right to get a better visual of the wing. Click "Save" when you are finished.

Wing Edition - xflr5 v6.57

Wing

☑ Symetric   ⦿ Right Side ○ Left Side   Insert before section 2 | Insert after section 2 | Delete section 2

Description:

	y ()	chord ()	offset ()	dihedral(°)	twist(°)	foil	X-panels	X-dist	Y-panels	Y-dist
1	0.000	0.200	0.000	0.0	0.00	Clark Y		13Cosine		19-Sine
2	1.000	0.140	0.060		0.00	Clark Y				

Wing Span   2.00 m
Area   0.34 m²
Projected Span 2.00 m
Projected Area 0.34 m²
Mean Geom. Chord 0.17 m
Mean Aero Chord 0.17 m
Aspect ratio 11.76
Taper Ratio   0.70
Root to Tip Sweep 2.58 °
Number of Flaps 0
Number of VLM Panels 494
Number of 3D Panels 1014

☑ Axes   ☑ Panels
☑ Surfaces   ☑ Outline
☐ Foil Names   ☐ Masses

Reset scale
Reset Mesh   Other ▼
Save | Discard

10. Next is designing the horizontal stabilizer. Click the "Define" button under the "Elevator" section.

11. Using the same process as the wing, please create a **horizontal stabilizer** with a the following parameters:

    a. NACA 0012 airfoil

    b. Span of 0.4 m (set the y() value to .2 m or 200 mm)

    c. Constant chord of 0.15 m (150 mm), no offset

12. Next, create a **vertical stabilizer** or "fin" with the following parameters:

    a. NACA 0012 airfoil

    b. Span of 0.2 m (200 mm)

    c. Root chord of 0.15 m (150 mm), tip chord of 0.075 (75 mm), taper ratio of 0.5

    d. Straight trailing edge, similar to the wing (offset of 75 mm)

13. Now, all of the main airfoils on the aircraft have been defined.

Note: If you accidentally exit the Plane Editor, click "Plane" > "Current Plane" > "Edit" to re-open the window and return to your plane editing.

14. Use the Plane Editor to adjust the physical locations of each airfoil compared to one another. Move the horizontal and vertical stabilizer to 0.70 m (700 mm) on the x-axis.

15. Finally, we will define the inertia of the aircraft. To begin, press the "Plane Inertia" button, and a new window should open.
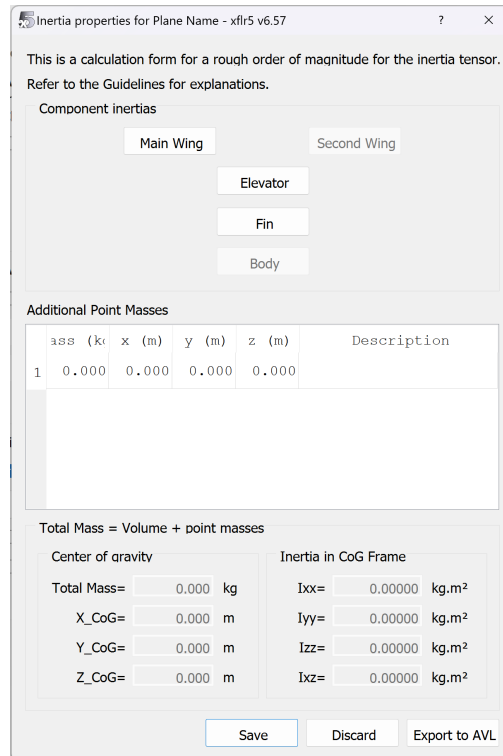
16. This screen is where the mass of the aircraft is defined. You can add individual point masses to simulate different components and sections of the aircraft. Begin by clicking on the "Main Wing" option, setting the "wing mass" to 0.7 kg, and clicking save.

17. Next, set the elevator's mass to 0.3 kg and the fin to 0.15 kg.

18. We will also add a point mass to the system to represent the fuselage and payload. On the bottom of the screen, where a table is shown labeled "Additional Point Masses," please create a mass of 1.5 kg located at (-0.2, 0.0, -0.06) meters or (-200, 0, -60) mm. Then click save to finish the inertia settings.

19. Finally, name your aircraft design and save your project.
   a. Click "Plane" > "Current Plane" > "Rename" and give your plane a fun name.
   b. Click "File" > "Save Project As…" and save your project.

Section 4: Aircraft Analysis

1. We will now run an analysis to compute lift, drag, and moment coefficients for the entire plane. Click "Analysis" > "Define an Analysis."
2. A new screen will appear on the screen. This new screen is where you will define the type of analysis you would like to run. Several options include a fixed speed, fixed lift, and fixed angle of attack. For this analysis, we will select a fixed speed and set the wind velocity to 15.0 m/s.
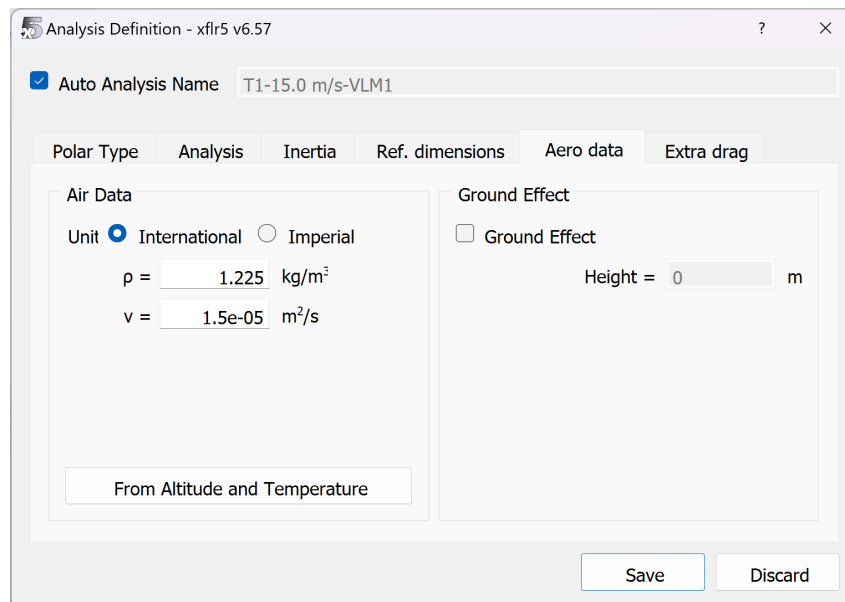
3. Please look through the other tabs for the controllable options. For now, we will leave these at default, but you may want to adjust these variables in your projects. When you are finished, click "Save."
4. On the right side of the screen is a new panel labeled "Plane analysis" and underneat that "Analysis Settings." For this analysis, we will vary the aircraft's angle of attack. Make sure the "Sequence" option is check, then set the starting angle at -5.0 degrees and the final angle at 15.0 degrees with 0.5 degree increments. Then select "Analyze."
5. Close the winow when the panel analysis is complete. There may be "Errors encountered". That's OK.
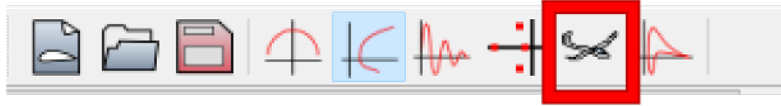
6. Select the icon with the airplane on the menu bar, as pictured above. This will alow you to visualize the aerodynamic forces acting on your plane.

    a. You can select the simulated angle of attack from the drop-down list in the "Object explorer" on the left.

    b. You can select which parameter to view via the display options on the right.

Deliverable 2: Display the Lift force at an angle of attack of 9 degrees. Orient the view so that you are looking at the yz-plane, with the x-axis coming out of the computer screen. Take a screenshot and upload it to the appropriate place on Canvas.

7. Experiment with the other possible parameters to display and view angles of the aircraft.

8. To view the information in a graph form, select the "Polar View" option pictured above. This will display the overall lift, drag, and moment coefficients for the entire aircraft configuration.

9. Select the various angles of attack in the Object explorer on the left. You should see a data marker appear at various locations on the graphs.

10. Focus your engineering awareness on the graph of Cm vs Alpha (pitch moment as a function of angle of attack). Note the slope is negative (stable). Select the angle of attack closest to trim, where $C_m = 0$. Write down the value of the trim angle of attack.

11. Focus your engineering awareness on the graph of CL/CD vs. angle of attack. Again, select the angle of attack that comes closest to the maximum lift-to-drag force ratio. Write down the angle of attack that maximizes the lift-to-drag ratio.

Deliverable 3: Take a single screenshot that includes all four plots. Upload it to the appropriate place on Canvas. Also, enter the values of the trim angle of attack and the angle of attack that maximizes the lift-to-drag force.

Section 5: Optimization

1. Note that the original trim angle of attack of the original design was negative. Adjust the parameters of your aircraft to a) make it positive and b) make it correspond to the maximum lift-to-drag force ratio.

2. Go to "Plane" > "Current Plane" > "Edit".

3. Recall that the intercept C_{m0} and trim angle of attack can be adjusted via the tail incidence angle i_t. That is, the angle that the tail is mounted to the fuselage. (See Example 2.1 in the textbook.)

    a. In the Plane Editor, adjust the "Tilt Angle" of the horizontal tail.

    b. Click the "Save" button in the plane editor.

    c. Click "OK" if a warning window about modifaction pops up.

4. Re-run the analysis for the new tail configuration.

    a. Select the original analysis in the Object explorer on the left.

    b. Click the "Analyze" button on the right.

5. Check the results with the new tail tilt angle. Is the trim angle of attack closer to the optimal CL/CD? Iteratively adjust the tail tile angle and re-run the analysis until you find the optimal value. Write down the optimal tail tilt angle.

Deliverable 4: Once you have optimized the design, select the trim angle of attack, such that the red marker shows its location on the plots. Take a single screenshot that includes all four plots. Upload it to the appropriate place on Canvas. Also, enter the value of the optimal tail tilt angle i_t.