Final Project
DUE MONDAY DEC 19, 5PM C.T. VIA CANVAS

Deliverables to Upload:

1. Final Report PDF: A professionally formatted report/technical paper with equations and embedded plots.

2. Any Matlab files used to generate your plots and analysis. Be sure to include anything needed to run all scripts, including files you may have downloaded from Piazza, or otherwise. Everything should be self-contained.

Grading Rubric

[20%] Report Formatting, Writing Style, and Presentation of Results. The report should be single-spaced, single column, with equations and plots integrated into the text, similar to any technical report or journal article. Plots and equations should be properly numbered (Figure 1, Table 1, Equation 1), and always referred to in the text. I suggest Overleaf (has many template options), or another LaTex text editor as an alternative to MS Word (although Word is OK). I do NOT recommend Google Docs for this assignment, as it is more difficult to format equations and embedded images. You should provide the proper sections and subsections in your report, and be sure to include references, cited appropriately. Plots should be with legible font size (at least 10 pt) proper axis labels, legends, and captions (do not use titles on plots). You do not need to include an abstract or introduction section. You will need to include a Summary and/or Conclusions section.

[30%] 2D Analysis Results and Discussion

[50%] 3D Analysis Results and Discussion

Part 0: XFLR5 Tutorial

Before beginning the project, please download and install XFRL5 `http://www.xflr5.tech/xflr5.htm`. Please complete the XFLR5 tutorial posted alongside the project description on Piazza, which was written by Alex Ames, specifically for Aerodynamics/Aerodynamics Lab. Please read thoroughly before starting the project. I will only accept Matlab plots of XFLR5 data (exporting data is included in the tutorial). XFLR5 is largely based on Professor Mark Drela's XFOIL code (`https://web.mit.edu/drela/Public/web/xfoil/`) which is a widely used design tool.

Part 1: 2D Analysis

In this portion of the project you will deal only with 2D airfoil sections.

First, consider a NACA 2412 airfoil section:

[a] **Thin Airfoil Theory**: Using thin airfoil theory, compute the lift coefficient, C_l as a function of α. Show your final expression and carefully explain the key steps in your analysis. Plot your final answer in Matlab.

[b] **Vortex Panel Method**: Using your vortex panel code, compute the lift and drag coefficient, and plot with respect to α. Plot C_l on the same figure as [a], clearly labeled with a legend. Plot C_d on its own figure. In the text be sure to show key equations used in this analysis. You will have to include your code when you submit the project.

[c] **XFLR5**: Using XFLR5, compute α vs. C_l and α vs. C_d. Export the data and plot on the same Matlab figures as above. Include a legend to clearly define which curve is which. You may have to make decisions on parameters for XFLR5, clearly state the parameters you chose, and the reasoning behind it.

[d] **Discussion Points:** Discuss your results for these three modeling strategies. Are they what you expected? What are the limitations for each of these techniques? How do they compare with experimental data? What would cause discrepancies between the various models, but also between the models and the experiments? How would you improve the models? This

may require some research into how XFLR5 computes 2D force coefficients, cite your sources appropriately. Experimental data available on the Piazza site from a previous assignment.

[e] **Mach Number Effect:** Using the Vortex Panel Method results, apply a compressibility correction to compute the α vs. C_l curve for $M = 0.1$, $M = 0.4$, and $M = 0.7$. Perform the same analysis in XFLR5 and include all three curves on the same plot (well-labeled). Be sure to show any important equations in your report.

[f] **Discussion Points:** Discuss your results for the Mach number dependence. Does XFLR5 use the same strategy for compressible corrections as we introduced in class? Does it produce similar results? (You may have to look at the documentation for XFOIL, which is the underlying algorithm in XFLR5)

[g] **Airfoil Properties:** Choose 3 different airfoil profiles described by the NACA 4-digit series. Explain your reasoning for choosing the profiles (e.g. all the same camber, with increasing thickness; same camber but different locations of max camber; same location of max camber but increasing camber, etc.) Compute the lift and drag polars at low ($M = 0.1$), moderate ($M = 0.4$), and high subsonic ($M = 0.7$) conditions. Directly compare the airfoils with one another (on the same plot) for each condition. Perform your analysis with the vortex panel method AND XFLR5. Discuss your results.

Part 2: 3D Analysis

In this section consider a finite-length wing with zero twist, comprised of a thin, symmetric airfoil section throughout the span.

First consider an elliptic wing with aspect ratio 6.

[a] **Lifting Line Theory Elliptic Solution**: Using lifting line theory described in the text, plot $\Gamma(y)/(2bV_\infty)$ *vs.* y/b, with α as a parameter (use α=2, 4, 6, 8, 10°). On two separate plots, compute α vs. C_L and α vs. C_{D_i} for the elliptic wing. Include the most important equations used for this theory, there is no need to repeat the entire analysis as it is described in the text (one can cite the textbook).

Next, consider a straight wing with constant chord length and aspect ratio 6. Use a Fourier sine series with 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 13, 14, 15, 16 terms to describe the lift distribution over the wing.

[b] **Lifting Line Theory General Solution**: Using Matlab, calculate the A_n coefficients of each series, in terms of α. Include in your report the tabulated the values of θ_0 you used to populate the matrix; and the results of A_n from above. Note the convergence of the A_n.

On a single figure, plot A_n/α *vs.* n, with N as a parameter ($2 \leq N \leq 16$) (i.e. n on the horizontal axis and A_n/α on the vertical axis). **NOTE:** you **ONLY** plot the points representing the A_n values. Do **NOT** connect them with lines.

For the 16-coefficient series, plot $\Gamma(y)/(2bV_\infty)$ *vs.* y/b, with α as a parameter (use α=2, 4, 6, 8, 10°).

Plot $\Gamma(y)/(2bV_\infty\alpha)$ *vs.* y/b, with N as a parameter ($2 \leq N \leq 16$).

Using the 16-coefficient series, calculate the total lift coefficient, C_L, and the lift-induced drag coefficient, C_{D_i}, both in terms of α.

Include all important equations and describe/discuss important conclusions from your analysis.

[c] **Discussion**: How do the elliptic and straight-bladed wings compare?

[d] **Taper Ratio**: Using XFLR5, add taper to the straight bladed wing, defined as the taper ratio $q = 1 - c_t/c_r$, where c_t is the chord length at the tip and c_r is the chord length at the root. Consider 5 values between (and including) $q = 0$ and $q = 0.9$, with $AR = 6$.

Compute δ and τ, make 2 separate plots: q vs. δ and q vs. τ.

Discuss the effects of taper ratio.

[e] **Effects of aspect ratio**: Using theory (not XFLR5), plot α vs. C_L for $q = 0$ with $AR = 3, 6, 12, 20$.

Discuss the effects of aspect ratio.