EAE 127 Applied Aircraft Aerodynamics

Project 4
Vortex Panel Method

Upload all files to the 'Assignments' section on Canvas in a single, compressed (zip) folder, which must contain a '.ipynb' Jupyter Notebook report (all Python code must run), a '.html' hard copy, and all data files necessary to run code. 'Run All' before uploading. (More details: 'EAE127_FAQ.pdf'). DUE: Monday 11/16/20 11:59pm

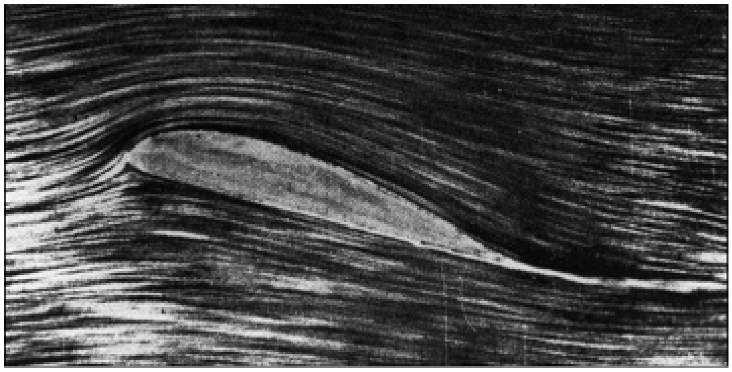

Fig. 1: Streamlines over an airfoil at angle of attack

1 Vortex Panel Method

You have been provided a vortex (lifting) panel method code. It is based on the 1st order method described in Anderson Section 4.10, where the additional constraint of the Kutta Condition is prevented from over-defining the system by leaving one panel out of the calculation of panel strength distributions, as described on Pg. 364 of Anderson's Fundamentals. You may use any part of the code for your own work, but simply running the demo will not be counted for credit.

To get familiar with the panel method code, solve for the $\mathbf{V_\infty = 1.5m/s, \rho_\infty = 1.2kg/m^3, c = 1m}$ flow over

a symmetric and cambered airfoil:	at zero and non-zero Angles of Attack:
NACA 0012	$\alpha = 0^o$
Selig 1223	$\alpha = 8^o$

and **plot flow streamlines** with the **airfoil geometry** for each solution (total of 4 plots). **Compare** the flow differences due to **airfoil geometry** and **angle of attack**.

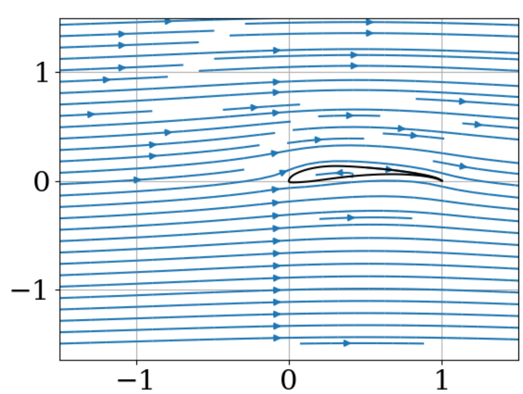

Fig. 2: Panel method solution streamlines for $\alpha = 0^o$ flow over a Selig 1223 airfoil

2 Calculating Lift

In computational aerodynamics, lift is typically calculated by integrating the surface pressure distribution of the discretized geometry. However, the Kutta-Joukowski theorem (Eqn 1) states that lift is also a function of the circulation Γ calculated by integrating the flow velocity along a closed contour C surrounding the geometry.

$$\begin{aligned} L' &= -\rho_\infty V_\infty \Gamma \\ \Gamma &= \oint_C V \cdot d\mathbf{s} \end{aligned} \tag{1}$$

Using your panel code, **calculate the coefficient of lift** C_l of a **NACA 0012 airfoil at angle of attack** $\alpha = 8^o$ by computing the **circulation** about **two**, closed, **independent** paths. (e.g. a circle and a square; NOT two circles)

Additionally, for each circulation integral, **plot the integration path and the airfoil geometry** as well as **flow streamlines** (you may use *streamplot* to achieve this, rather than plotting the velocity vectors on the integration path as shown below).

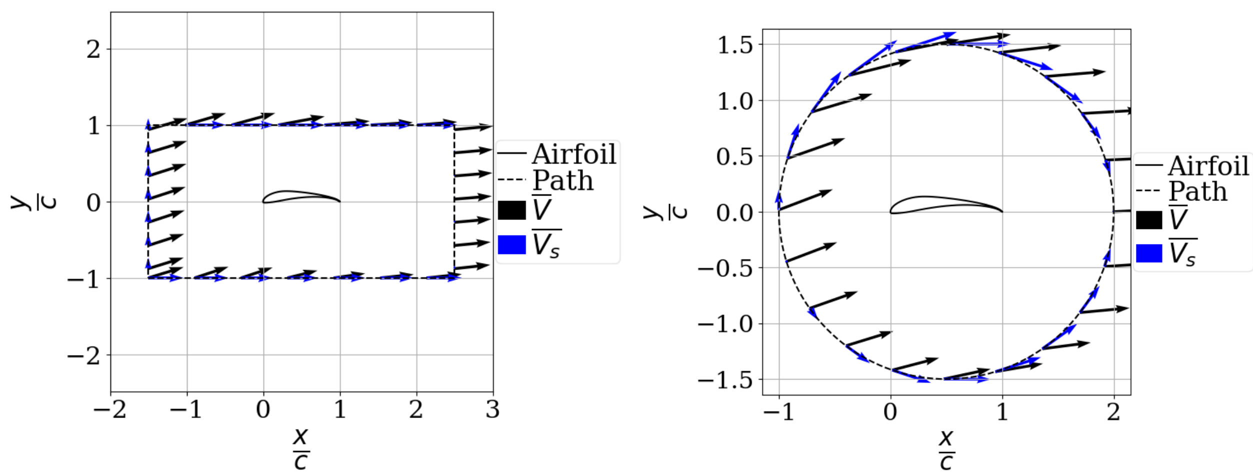

Fig. 3: Example of Kutta-Joukowski theorem lift solution for rectangular and circular integration paths (Selig 1223 at $\alpha = 11^0$)

Compare the results of **each method** to each other and to **inviscid XFOIL** results **in a single table**. (Get airfoil geometry coordinates from XFOIL).

3 Lift Curve

Calculate the lift curve ($C_l vs. \alpha$) for the NACA 0012 airfoil using the following techniques:

Inviscid: Thin airfoil theory
Inviscid: Your vortex panel method and Kutta-Joukowski lift theorem (technique used in Problem 2)
Viscous: XFOIL/pyxfoil, with $Re = 5e6$

For each lift curve, angle of attack must range from 0^o to at least 25^o, and you must calculate enough points in between to give an accurate representation of each curve's shape.

Co-plot the three lift curves (same figure) and **compare and explain the differences**. **What causes the viscous case to noticeably diverge** from the inviscid results at higher angles of attack?

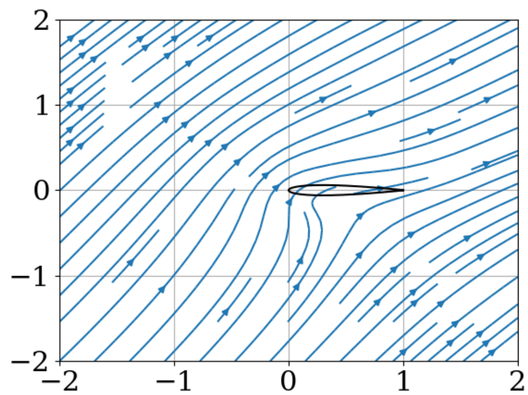

Fig. 4: Panel method solution streamlines for $\alpha = 45^o$ flow over a NACA 0012 airfoil

Finally, **plot flow streamlines** with the **airfoil geometry** for the NACA 0012 at angle of attack $\alpha = 45^o$ (like Fig. 4 above). **Where is the leading edge stagnation point located**, approximately? Does this solution seem realistic? How would the **streamlines change for viscous flow**?

4 Additional Problems

4.1 Lifting Rotating Cylinder

Let's build a rotor-wing! Assume the same weight, wingspan, and velocity of a Cessna 172 light aircraft at sea level conditions. Assume dynamic equilibrium, and plot cylinder rotational velocity Ω, vs cylinder radius R.

4.2 Airfoil Performance

Calculate the lift and drag forces for three different 12% thick NACA airfoils (2412, 4412, and 23012) at $\alpha = 5^o$ and two $Re_c = 3E6, 9E6$. Use Appendix D of Anderson's Introduction to Flight. Analyze for 2D airfoil conditions. Assume a chord of $7.0ft$, and flight conditions of Standard Atmosphere at $5000ft$ (you may assume changes in μ between flight conditions are negligible). Report your answers with the appropriate units.

4.3 Airfoil Efficiency

Estimate the max L/D for the same three airfoils, at $Re_c = 9E6$.

4.4 Induced Drag

Calculate the induced drag force of an airplane of weight $9000N$. Assume wing area $S = 18.3m^2$, aspect ratio $AR = 9$, and span efficiency=0.75. Consider two cases: at cruise (V=245km/hr, $h = 32000ft$), and during landing ($V = 85km/hr$, $h = 0ft$). Use altitude tables to determine necessary freestream parameters.

4.5 Airfoil Selection

What is your favorite airfoil? (Any airfoil, not just of the ones used in this project). **Why**? (If you'd like to, you can plot your airfoil's geometry to help you explain).