EAE 127 Applied Aircraft Aerodynamics

Project 2
Potential Flow

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 11/2/2020 2:00pm

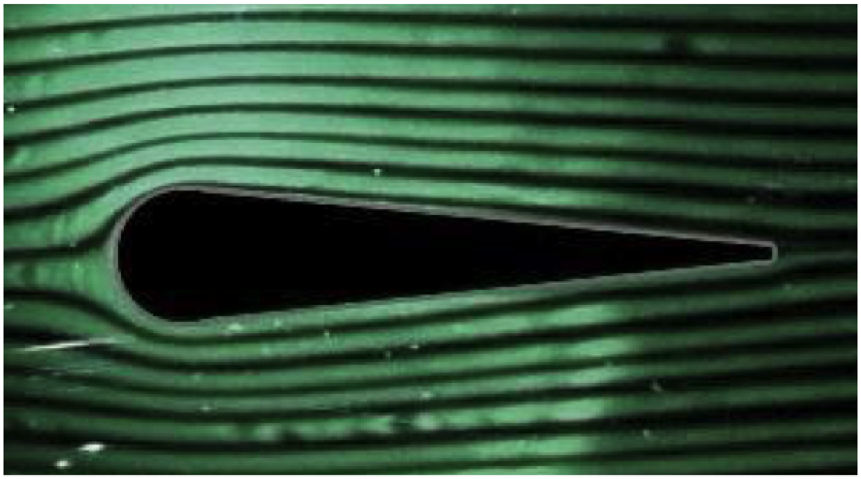

Fig. 1: Streamlines over an airfoil at angle of attack

1 Freestream/Source Superposition

Derive the equations for the stagnation point and dividing streamline of a source in freestream flow, starting with Eqn. 3.75 in Anderson **(for practice with LaTeX)**. Next, create a code to simulate the superposition of uniform flow and a potential flow source (Freestream velocity $V_\infty = 0.6\,{}^{m}/_{s}$).

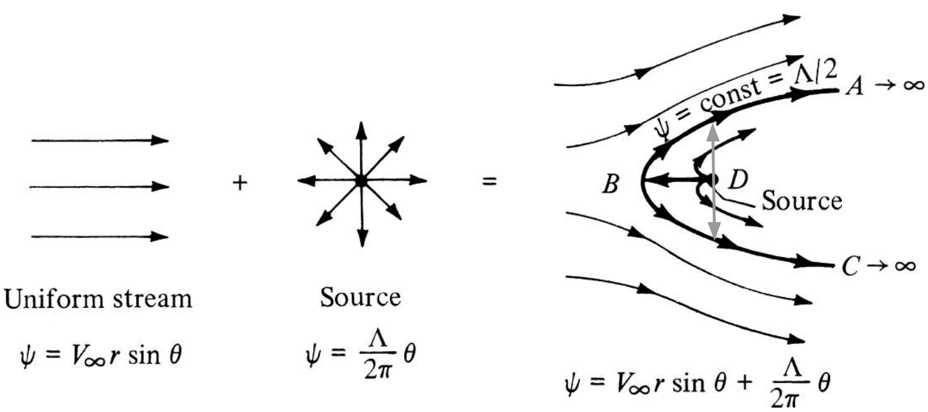

Fig. 2: Freestream+source superposition showing dividing streamline *(Source: Anderson)*

For a source strength of $\Lambda = 2.1$, **plot and label** (in a single figure):

- **Streamlines**
- **Velocity potential contours** (can be on separate plot)
- **Dividing streamline**
- **Dividing streamline diameter** D (gray arrow in Fig. 2) and its **length in** *meters*
- **Source location**
- **Stagnation point location**

(Reference AeroPython Lesson02 by Lorena Barba for help in setting up your potential flow code). Use the *streamplot* function to plot streamlines and the *contour* function to plot the velocity potential. The *plt.annotate* function can be handy for annotating points with text. The Jupyter Notebook *%%capture* command can be used at the beginning of a code cell to 'capture' incremental plotting code without showing the incomplete plot (the final plot can be shown later with a different code cell).

Next, **plot the dividing streamline diameter** as a function of source strength, and **describe this relationship.**

2 Potential Flow Airfoil Representation

In potential flow, any streamline can be thought of as a solid body, since no flow will cross its boundary, by definition. **Approximate the first three quarters of the geometry** (from $x = 0$ to $x = 0.75c$) of the symmetric **NACA 0020** for:

- $\alpha = 0^o$ (non-lifting flow)
- $V_\infty = 0.5 \, {}^{ft}\!/_{s}$
- Chord length $c = 3ft$

using a distribution of **ten or more sources and sinks** along the chord line.

Select singularity strengths and locations such that the dividing streamline closely resembles the airfoil surface geometry (**ensure that the sum** of the strengths of all of the sources and sinks **is zero** so that the dividing streamline is a closed body). Use trial and error to find the best singularity distribution (no need to solve a system of linear equations, this is not a panel method ... yet).

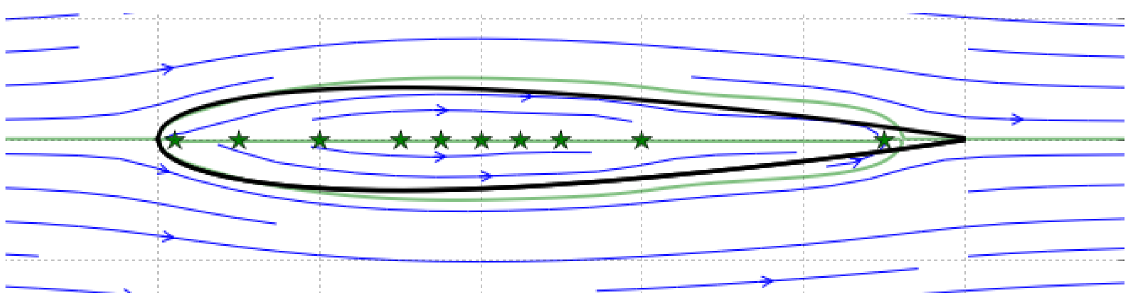

Fig. 3: Source distribution (green stars) dividing streamline approximation (green line) of NACA symmetric airfoil (black)

For <u>at least three</u> of your trials of singularity distributions, **plot in a single figure:**

- **Actual airfoil geometry**
- **Dividing streamline of the source distribution**
- **Singularity locations**

Quantify the error of the approximation of **each trial** using a metric based on the sum of the magnitude of the difference between the actual NACA equation geometry and the dividing streamline profile, such as the one below:

$$arbitrary\_error\_metric = \frac{1}{nx} \sum_{x=0}^{\frac{3}{4}c} \left| z_{div}(x) - z_{geom}(x) \right|$$

where z_{div} and z_{geom} are the vertical-axis geometry coordinates of the dividing streamline and NACA geometry, respectively, and nx is the number of points used in the sum (normalizes our metric for "apples to apples" comparisons).

Discuss the advantages and limitations of this method for approximating flow over an airfoil. **How might this method be improved** to produce more accurate results?

3 Source Wall

Approximate $0.5 ^{ft}/_{s}$ uniform flow normal to a finite wall using offset vertical distributions of $\mathbf{N} = \mathbf{5}, \mathbf{11}, \mathbf{101}$ sources and sinks (see Fig. 4).

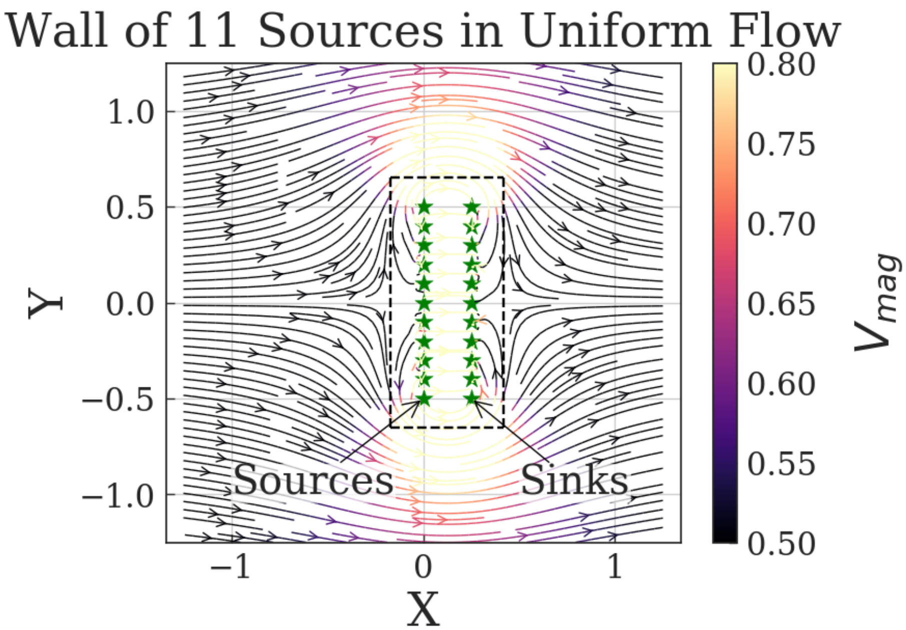

Fig. 4: A vertical "wall" (represented by dashed lines) created by equal an opposite columns of sources and sinks.

To create the wall with potential flow elements, make a vertical distribution of sources $1\,ft$ high, followed by a $1\,ft$ high vertical distribution of equal and opposite strength sinks placed $0.25\,ft$ downstream of the sources. Let the strengths of all singularities be equal to $\Lambda = \pm\frac{5}{N}$, where N is the number of sources/sinks in each vertical column.

For each distribution, **estimate the shape of the wall** created by the singularities using dashed lines. **Report the height and width** of each wall and **compare and justify any differences**. What differences are there **between this simulation and real (viscous) flow** past a normal wall?

4 Additional Aerodynamics Problems

Answer the following textbook problems in your Jupyter Notebook, writing out your derivations, showing your work in code, and discussing your results.

4.1 Center of Pressure

Consider an NACA 2412 airfoil (the meaning of the number designations for standard NACA airfoil shapes is discussed in Chapter 4). The following is a tabulation of the lift, drag, and moment coefficients about the quarter chord for this airfoil, as a function of angle of attack.

α [deg]	C_l	C_d	$C_{m,c/4}$
-2.0	0.05	0.006	-0.042
0	0.25	0.006	-0.040
2.0	0.44	0.006	-0.038
4.0	0.64	0.007	-0.036
6.0	0.85	0.0075	-0.036
8.0	1.08	0.0092	-0.036
10.0	1.26	0.0115	-0.034
12.0	1.43	0.0150	-0.030
14.0	1.56	0.0186	-0.025

From this table, **plot** in your Jupyter Notebook the variation of x_{cp}/c as a function of α and **discuss the trends**.

4.2 Dynamic Similarity

Consider two different flows over geometrically similar airfoil shapes, one airfoil being twice the size of the other (chord is twice as long). The flow over the smaller airfoil has freestream properties given by:

$T_\infty = 199K$, $\rho_\infty = 1.23 kg/m^3$, and $V_\infty = 141 m/s$.

The flow over the larger airfoil is described by:

$T_\infty = 400K$, $\rho_\infty = 1.739 kg/m^3$, and $V_\infty = 200 m/s$.

Assume that both μ and a are proportional to $T^{\frac{1}{2}}$. **Are the two flows dynamically similar?**

4.3 Similarity Parameters

Consider a Lear jet flying at a velocity of $200 m/s$ at an altitude of $10 km$, where the density and temperature are $0.414 kg/m^3$ and $223K$, respectively. Consider also a one-fifth scale model of the Lear jet being tested in a wind tunnel in the laboratory. The pressure in the test section of the wind tunnel is $1 atm = 1.01 \times 10^5 N/m^2$. **Calculate the necessary velocity, temperature, and density** of the airflow in the wind-tunnel test section such that the lift and drag coefficients are the same for the wind-tunnel model and the actual airplane in flight.

4.4 Lift Coefficient

Estimate the **coefficient of lift** (**C_L**) for a Boeing 787 at maximum gross weight for:

a. Cruise at $42 kft$ on a standard atmosphere day
b. Landing in Denver on a standard atmosphere day
c. Landing in San Francisco on a standard atmosphere day

Estimate the **coefficient of lift** for a Cessna 152 at maximum gross weight for:

d. Cruise at $7 kft$ on a standard atmosphere day