EAE 127 Applied Aircraft Aerodynamics

Project 1
Introduction to Airfoils and XFOIL

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 10/26/20 2:00pm

Fig. 1: General aviation aircraft demonstrating wing cross-section

This project will delve deeper into the concept of airfoils. An airfoil is defined as the cross-sectional shape of an infinite wing (imagine in Fig 1 above if the wing tip facing you extended infinitely out of the page as well as infinitely into the page on the other side. We then slice into this wing on a plane parallel to the page and observe the flow surrounding this cross-section).

Since the wing extends to infinity in both directions, there are no tip effects and, thus, no variations in the flow in the spanwise direction y. We, therefore, refer to this kind of flow as 2-dimensional, since the flow only varies in the x (chordwise) and z (vertical) directions.

1 Airfoil Geometry Characteristics

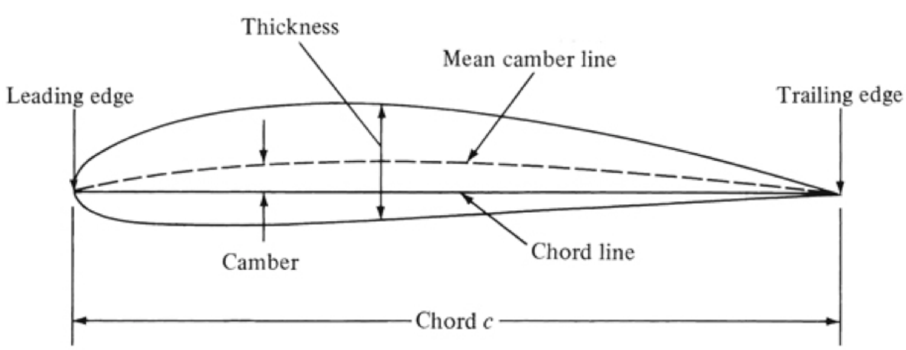

Fig. 2: Airfoil geometry nomenclature (*Source: Anderson Introduction to Flight, Figure 5.3*)

We will now practice plotting some different airfoils similar to Fig. 2 above. We will plot two airfoils from the NACA airfoil series, which are airfoil geometries can can be created entirely from an equation and depend only on a few global parameters like maximum thickness and maximum camber height.

For each of the following airfoil plots:

- use a chord length of $c = 1$
- **plot the camber line as a dashed line**
- **label the upper and lower surface separately, as well as the camber line** in a legend
- **use equal axes** when plotting geometries

1.1 Symmetric Airfoils

Referencing Fig. 2 above (which is a cambered airfoil), we see that the airfoil's thickness is defined as being equally distributed in the vertical direction on either side of the camber line. If that camber line is perfectly horizontal (no camber) and coincident with the chord line, we end up with an airfoil that is perfectly reflected about the chord line, which we call a symmetric airfoil.

The NACA thickness distribution equation is:

$$\frac{z_t}{c} = \frac{t}{0.2}\left[0.2969\sqrt{\frac{x}{c}} - 0.1260\left(\frac{x}{c}\right) - 0.3516\left(\frac{x}{c}\right)^2 + 0.2843\left(\frac{x}{c}\right)^3 - 0.1015\left(\frac{x}{c}\right)^4\right]$$

where z_t is the thickness on <u>either side</u> of the camber line (half the total thickness at that point) and t is the ratio of maximum airfoil thickness τ to chord length c ($t = \tau/c$).

The last two digits in a NACA 4-digit code correspond to the thickness ratio t. **Plot a NACA 0018 symmetric airfoil** (camber line is chord line and $t = 0.18$).

1.2 Cambered Airfoils

Now that we have seen a symmetric airfoil, let's look at the NACA 0018's cambered cousin, the NACA 2418. Just like the previous case, the maximum thickness ratio $t = 0.18$, but now the first two numbers are no longer zero, which means the camber line will not be horizontal.

The first digit ends up corresponding to the ratio of maximum camber to chord length $m = 0.02$ and the second digit corresponds to the chord-wise location of maximum camber non-dimensionalized by chord length $p = 0.4$. Like before, we will plot the same thickness distribution about the camber line, but the camber line will now vary according to the following equation:

$$\frac{z_c}{c} = \begin{cases} \dfrac{m}{p^2}\left(2p\left(\dfrac{x}{c}\right) - \left(\dfrac{x}{c}\right)^2\right), & 0 \le x \le pc \\[2ex] \dfrac{m}{(1-p)^2}\left((1-2p) + 2p\left(\dfrac{x}{c}\right) - \left(\dfrac{x}{c}\right)^2\right), & pc \le x \le c \end{cases}$$

Plot the NACA 2418 geometry as we did in the previous problem, and **comment on the differences and similarities** between the two airfoils.

2 Airfoil Wake Drag

Now that we understand a bit more about airfoil terminology, we can start to talk about what really matters in 2D analysis: flow-induced forces and moments. A typical aerodynamic analysis involves first characterizing the flow about our geometry, either through an experiment or a simulation, then using this flow information to calculate its effect on our geometry using a variety of methods.

For this problem, we will employ a technique commonly used to calculate airfoil drag from wind tunnel test data called the wake momentum deficit method, which is detailed in Anderson Fundamentals of Aerodynamics, Section 2.6. This section derives the below equation, which states that in incompressible flow ($\rho = const.$), the sectional drag D' (drag per unit span along the infinite wing) is equal to the integral of the momentum difference between the freestream flow in front of the airfoil and the wake after the airfoil.

$$D' = \rho \int_h^b u_2(u_1 - u_2)dy$$

Anderson Equation 2.84, where h and b are the bottom and top of the control volume drawn around the airfoil in Fig. 3, respectively.

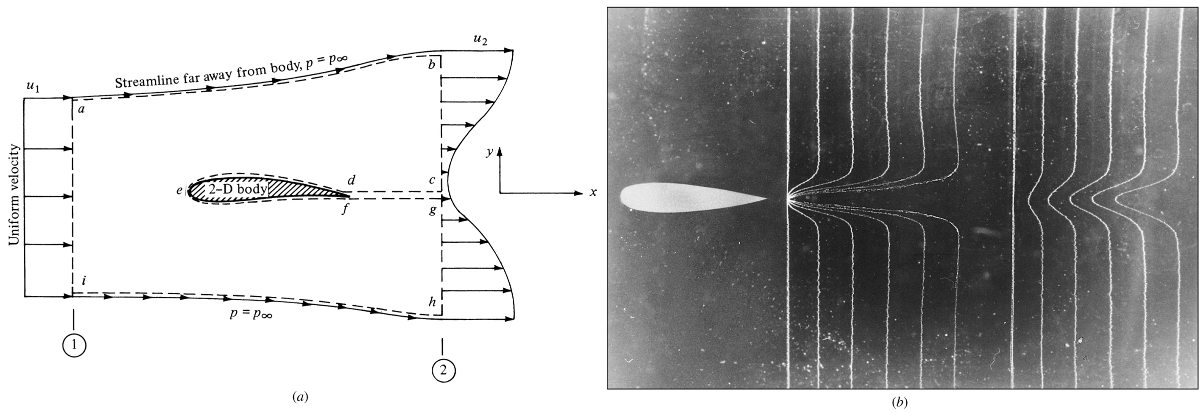

Fig. 3: Airfoil wake velocity distribution (*Source: Anderson Fundamentals of Aerodynamics, Figure 2.20*)

Compute the sectional drag D' of a wing by applying the momentum deficit equation to the provided wake data in the file "WakeVelDist.dat", which is comma delineated into three columns: y/c, u_1, and u_2. Assume air is incompressible with $\rho = 1.2kg/m^2$.

You will notice that there are only a few data points: wake data was recorded at only a few discrete locations within the continuous wake. **Improve your integration results by fitting the wake data (u_2) with a polynomial** of appropriate order. **Plot the original wake data and your polynomial fit**, with y/c on the vertical axis and u_2 on the horizontal axis.

3 XFOIL Introduction

XFOIL is a code developed by Mark Drela at MIT that simulates how air flows over an airfoil (a 2D cross-section of an infinite wing). It uses an inviscid panel method code, where the airfoil body is broken up into a finite number of linear panels and flow is constrained to have streamlines tangent to each of these panels. Solutions from this potential flow method are used in conjunction with boundary layer formulations to provide a viscous flow estimation as well. Thus, XFOIL gives us a computationally non-intensive method of calculating the flow around an airfoil and consequently the forces and moments induced on the airfoil by this flow. XFOIL can be run on your own computer after downloading the source code:

`http://web.mit.edu/drela/Public/web/xfoil/`

XFOIL is a command-line program and is controlled by a set of text commands. Remembering command names and navigating through text menus can be tedious and confusing, so, for your convenience, the previous TA Logan Halstrom has developed a Python-based XFOIL automation script called 'pyxfoil.py'. This script will allow you select or load airfoil geometries, specify Reynolds numbers and Angles of Attack, and save force coefficient and surface pressure distribution data to file all via a Python script. 'pyxfoil' is available to you on Canvas, and you may use it as you see fit for any application in this course as well as outside of class.

In this project, we will use XFOIL to observe the flow and force differences between symmetric and cambered airfoils.

For the **two airfoils** in Problem 1:

- NACA 0018
- NACA 2418

use XFOIL to simulate the surrounding flow for:

- Inviscid flow **Re = 0** (Equivalent to $Re = \infty, \mu = 0$)
- Viscous flow **Re = 6e5**

and two angles of attack α:

- $\alpha = 0^o$
- $\alpha = 11^o$

(A total of **4 cases for each airfoil**). For each case, **tabulate the lift coefficient C_l, the drag coefficient C_d, and the moment coefficient C_m**, and **save the pressure distribution to a file**.

3.1 Symmetric vs Cambered Airfoil Surface Pressure Comparison

After creating your data, make two plots (one for each angle of attack α) of the **surface pressure distributions C_P *vs.* x/c** for the viscous case only ($Re = 6e5$). **Plot distributions for both airfoils** on each plot and **label the upper and lower surface pressures separately**. (You will have 4 separate lines on each plot).

To make the plots more intuitive (with the upper surface pressure on the upper portion of the graph like Fig. 4 above) **reverse the direction of the y-axis** using the command "*plt.gca().invert_yaxis()*".

Discuss the plots, noting the **differences between the two airfoils** at a given α as well as the **effect of angle of attack α on the shape of the surface pressure distribution**. Look at the y-axis scales of each plot: **which plot has higher peak C_P and why?**

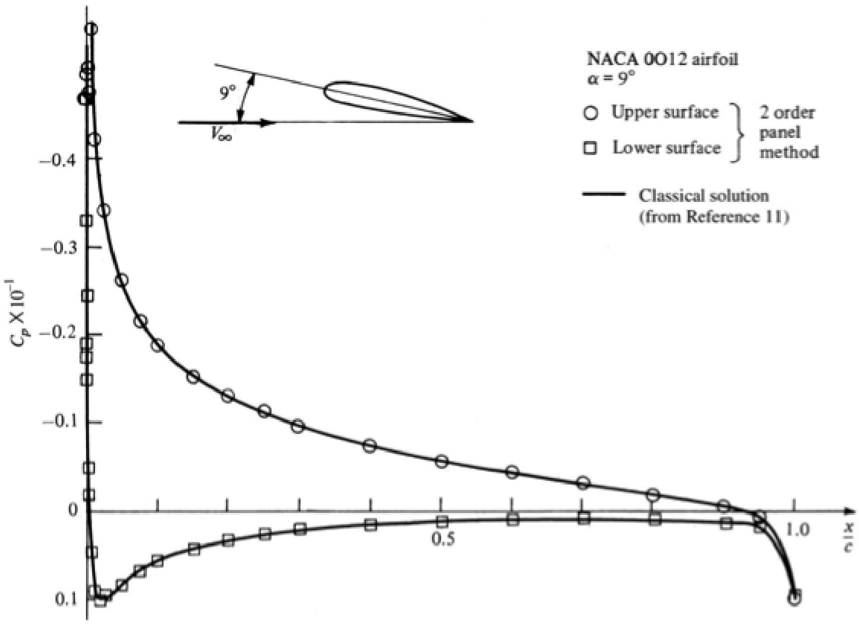

Fig. 4: A typical surface pressure distribution plot (Note that the y-axis have been **REVERSED** so that it runs from positive to negative)

3.2 Inviscid vs Viscous Flow Comparison

Compare the differences between viscous and inviscid flow by **plotting the surface pressure distributions** for the NACA 0018 at $\alpha = 11^o$ for both Reynolds numbers on the same plot, using the same plotting guidelines as before.

Note the differences in the pressure distributions and **suggest reasons why these differences occur**.

3.3 Force and Moment Integration

Though XFOIL has already done the work for you, we will now practice our numeric integration skills to calculate the force and moment coefficients of the airfoils. Using the surface pressure data you just plotted for the **NACA 2418** at $\boldsymbol{\alpha = 11^o}$, calculate the **inviscid** $Re = 0$ **and viscous** $Re = 6e5$ **lift coefficients** C_l**, the drag coefficients** C_d**, and the moment coefficients** C_m using numeric integration:

$$C_n = \frac{1}{c}\left[\int_0^c (C_{P,l} - C_{P,u})dx\right]$$

$$C_a = \frac{1}{c}\left[\int_0^c \left(C_{P,u}\frac{dz_u}{dx} - C_{P,l}\frac{dz_l}{dx}\right)dx\right]$$

$$C_{m,LE} = \frac{1}{c^2}\left[\int_0^c (C_{P,u} - C_{P,l})x dx\right]$$

and coordinate rotation:

$$C_l = C_n \cos\alpha - C_a \sin\alpha$$

$$C_d = C_n \sin\alpha + C_a \cos\alpha$$

Use a chord length of $c = 1$ to integrate with non-dimensional $\frac{x}{c}$.

For the axial force coefficient C_a, you will need to compute the airfoil surface slope $\frac{dz}{dx}$ using finite differencing. This can get complicated at the endpoints, so I suggest using Numpy's "*gradient*" function on each upper and lower surface, which computes the central difference on interior points and uses forward and backward differencing for the end points, all with second-order accuracy.

Tabulate C_l, C_d, C_m for the NACA 2418 for **both Reynolds numbers** (2 tables total), presenting both your integration results and XFOIL's results, according to the following format:

	C_l	C_d	C_m
XFOIL	$C_{l,xf}$	$C_{d,xf}$	$C_{m,xf}$
Integration	$C_{l,int}$	$C_{d,int}$	$C_{m,int}$

To better match the values of C_m, you will need to apply the Moment Transfer Theorem to compute the moment about the quarter chord, which is the value returned by XFOIL.

$$C_{m,c_{\frac{1}{4}}} = C_{m,LE} + C_l x_{c_{\frac{1}{4}}}$$

where $x_{c_{\frac{1}{4}}} = 0.25$ for $c = 1$.

Discuss how well your integrated values compare to XFOIL's and hypothesize the causes of any differences.

4 Additional Aerodynamics Problems

Answer the following textbook problems in your Jupyter Notebook, showing your work and results:

4.1 Anderson problem 1.15 (page 100)
4.2 Anderson problem 3.11 (page 310)
4.3 Anderson problem 3.16 (page 310)