Airfoil Aerodynamics

Reading: pp. 408–439 (Fox, Pritchard and McDonald)

Objectives

(i) measure the lift and drag forces on a model NACA 0012 airfoil using a lift/drag balance,

(ii) measure the static pressure along the top and bottom surfaces of the airfoil, and

(iii) understand how the lift, drag, and static pressure varies with angle of attack under subsonic flow conditions.

Background

In 1915, the U.S. Congress created the National Advisory Committee for Aeronautics (NACA), the predecessor to NASA, in order to accelerate American aviation technology, which surprisingly was beginning to lag behind Europe. In the late 1920's and early 1930's, NACA designed and tested a series of airfoils in the variable density wind tunnel located in the Langley Memorial Aeronautical Laboratory, Norfolk, Virginia. Each airfoil was designated by a four digit number that represented the geometry of the airfoil. After the mid 1930's, airfoil geometries became more sophisticated, as did the numbering scheme, in order to optimize lift while simultaneously decreasing drag.

Airfoil Geometry

In the present experiment, we consider the aerodynamic performance of a NACA 0012 airfoil, shown in Figure 1. The lift force (F_L) acts perpendicular to the direction of the approach flow; while, the drag force (F_D) acts parallel to the approach flow. In general, the chord line defines the straight line joining the leading edge and trailing edge. The length of this line is referred to as the *chord* (c). The *angle of attack* (α) is defined as the angle between the approach flow and the chord line. The maximum distance between the upper and lower airfoil

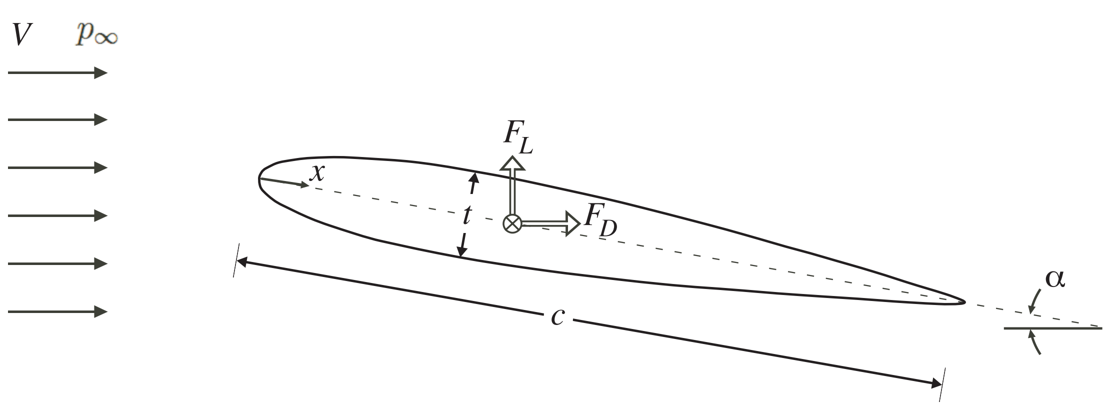

Figure 1: Schematic of the NACA 0012 airfoil at an angle of attack α.

surfaces is called the *thickness* (t). The mean camber line defines the curve of points that lie halfway between the upper and lower airfoil surfaces. The maximum distance between the mean camber line and the chord line is referred to as the *camber*. The shape of the airfoil, when viewed from above looking down, is called the *planform*. The planform area (A_p) is the projected area of the planform, i.e., the area of the shadow cast by a light directly above the airfoil when the airfoil is at zero angle of attack. The *span* (s) is the distance (into the page as viewed in Figure 1) from one wing tip to the other. The aspect ratio (AR) measures how long and slender the airfoil is from wing tip to wing tip. For a rectangular wing, $A_p = s \cdot c$ and $AR = s^2/A_p = s^2/(s \cdot c) = s/c$.

As indicated in Figure 1, the NACA 0012 airfoil is symmetric about the chord line and has zero camber. The naming convention (last two digits) indicates the ratio $t/c = 0.12$. The maximum thickness occurs at $x/c = 0.3$, where x is the distance along the chord line as measured from the leading edge. In the present experiment, the airfoil spans the entire width of the wind tunnel; therefore, s is equal to the width of the wind tunnel test section. Since the airfoil is rectangular (as viewed from the top looking down), c does not vary across the span.

Aerodynamic Performance

The Reynolds number for airfoils is based on the chord length and approach flow speed,

$$Re_c = \frac{V\,c}{\nu}, \tag{1}$$

where ν denotes the kinematic viscosity of the fluid. Depending on the Reynolds number, the boundary layer that develops over the airfoil may be laminar or turbulent. For flow over a flat plate, the transition from laminar to turbulent flow generally occurs near $Re_x \approx 5 \times 10^5$, where x is the distance along the plate as referenced to the leading edge. The coefficients of lift (C_L) and drag (C_D) for airfoils are defined as

$$C_L = \frac{F_L}{1/2\,\rho\,V^2\,A_p} \qquad \text{and} \qquad C_D = \frac{F_D}{1/2\,\rho\,V^2\,A_p}. \tag{2}$$

The quantity $(1/2\,\rho\,V^2)$ denotes the *dynamic* pressure associated with the approach flow. In the case of the NACA 0012 airfoil section using in the present experiment, the planform area is given by $A_p = s \cdot c$.

The coefficient of pressure (C_p) is defined as

$$C_p = \frac{p - p_\infty}{1/2\,\rho\,V^2}, \tag{3}$$

where p is the *static* pressure along the airfoil surface and, hence, is a function of x. The quantity p_∞ denotes the *static* pressure of the approach flow. The location of the *minimum* C_p gives some indication of where the boundary layer transitions from laminar to turbulent. When this happens, the skin friction drag increases dramatically. Therefore, from a design standpoint, it is desirable to maintain laminar flow along the airfoil and push the transition point farther aft (in terms of x/c).

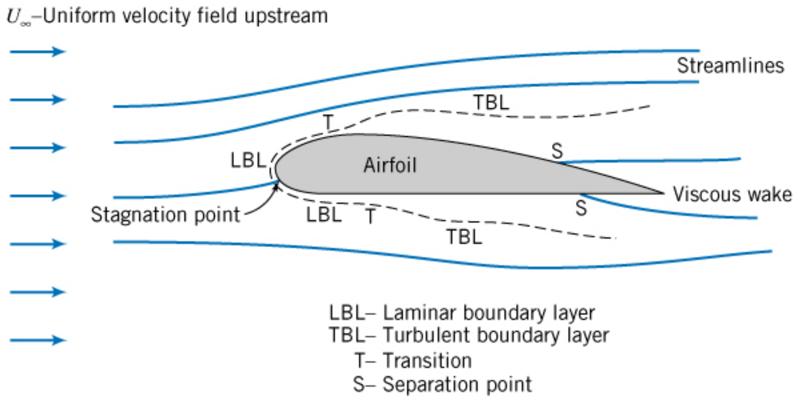

Figure 2: Illustration of the flow phenomena for an airfoil near zero angle of attack.

Figure 3: Flow visualization images of an airfoil at zero angle of attack (left) and near stall (right).

Figure 2 illustrates the different flow phenomena that exist for the case of airfoils. All three performance parameters: C_L, C_D, and C_p vary with both angle of attack α and Reynolds number Re_c. As the angle of attack increases, the separation point (S in Figure 2) moves upstream toward the leading edge of the airfoil. At a high enough angle of attack, the lift force will drop drastically due to the fact that the flow has separated along the entire chord of the airfoil, as shown by the flow visualization image in Figure 3. The term for this condition is *stall*. One of the objectives of this experiment is to gain some understanding of this behavior for the relatively simple case of the NACA 0012 airfoil.

Laboratory Tasks

Experiments are conducted on a NACA 0012 airfoil in a wind tunnel with a 12 inch $\times$ 12 inch cross section. The chord length is $c = 4$ inches, the thickness is $t = 0.48$ inches ($t = 0.12 \cdot c$), and the span is $s = 12$ inches. The airfoil is mounted horizontally in the wind tunnel, and extends the entire width of the test section. Two different NACA 0012 airfoils will be used: one that mounts directly to the lift/drag balance to allow measurement of the lift and drag coefficients, and the another that has small pressure taps along the centerline of the airfoil surface to allow measurement of the coefficient of pressure. Both airfoils pivot thereby permitting a range of angle of attacks to be investigated. The approach flow V may be varied by controlling the power supplied to the wind tunnel fan.

IMPORTANT: Static pressure measurements will be obtained using the pressure transducer. There are two ports, labeled "total" and "static", located on the instrument box. To measure the static pressure along the top surface of the airfoil, for example, you need to connect the appropriate tube to either port and leave the other port open to atmosphere. Depending on the configuration, a *positive* or *negative* reading may be seen on front display of the transducer. The actual pressure on the airfoil's top surface should be negative (less than atmospheric). Hence, you will need to manually assign the correct sign for the static pressures in your MATLAB code. The approach flow static pressure should also be negative because the test section is in vacuum due to the fan sucking instead of blowing the air through the tunnel.

The lab procedures are outlined below.

1. Determine the air density:
   (a) Measure the room temperature using the thermometer.
   (b) Measure atmospheric pressure using the mercury barometer.
   (c) Use the ideal gas law to calculate the local air density.
   (d) Use equation (A.1) in the textbook to determine the dynamic viscosity based on the measured air temperature.
2. Obtain lift and drag measurements:
   (a) Mount the lift/drag balance to the bottom of the wind tunnel such that the sting extends into the test section.
   (b) Connect the cable from the lift/drag balance to the port labeled "Dynamometer" on the back of the instrument box.
   (c) Secure the airfoil to the sting of the lift/drag balance by sliding the pivot (located on the underside of the airfoil) over the top of the sting. Use the cap screw provided to pin the airfoil pivot to the sting.
   (d) Set the angle of attack of the airfoil to $\alpha = 0$ deg. Use the set screw to tighten the airfoil in this configuration.
   (e) Zero the lift/drag balance:
      i. Switch the center dial on the front panel of the instrument box to "drag". Turn the brass thumbwheel for "drag", located on the lift/drag balance, until the display on the front panel of the instrument box reads "0". NOTE, the thumbwheel for "drag" is the smaller of the two thumbwheels.
      ii. Switch the center dial on the front panel of the instrument box to "lift". Turn the brass thumbwheel for "lift", located on the lift/drag balance, until the display on the front panel of the instrument box reads "0".
   (f) Set the fan power between 40–50 Hz.
   (g) Measure the approach flow speed:

i. Position the pitot-static probe at the test section entrance. Traverse the pitot-static probe in the vertical direction until it is about 1.5 inches from the ceiling of the test section. This will minimize the flow disturbance of the pitot-static probe on the airfoil.

ii. Connect the pitot-static probe to the "total" and "static" pressure ports on the pressure transducer.

iii. Take ~30 seconds of data of the *dynamic* pressure using the provided labview vi. Save the file with an appropriate name as in previous labs. Make sure to note the correct units.

(h) Measure the lift force by switching the center dial on the front display of the instrument box to "lift" and record the reading on the display. Measure the drag force by switching the center dial to "drag" and record the reading. Make sure to write down the correct conversion factor between volts on the display and actual force units (this will be provided by the TA).

(i) Adjust the angle of attack and record the lift and drag measurements. Repeat for angles of attack between 0–16 deg.

3. Obtain static pressure measurements along the airfoil surface:

(a) Remove the airfoil used for the lift/drag measurements by removing the pivot screw and sliding the airfoil out through the side access hole in the wind tunnel.

(b) Unscrew the lift/drag balance from the underside of the wind tunnel and seal the opening with the matching cover.

(c) Slide the airfoil with the pressure taps through the side access hole and position it horizontally across the test section of the wind tunnel. Make sure the airfoil is oriented such that the pressure taps are located on the top surface of the airfoil.

(d) Set the fan power to the same setting as used in the lift/drag experiment.

(e) Measure the *dynamic* pressure of the approach flow with the pitot-static probe, using the same procedure as outlined above.

(f) Measure the *static* pressure of the freestream flow at an x location near the leading edge of the airfoil, using the pitot-static tube by removing the total pressure tube connected to the port of the transducer. Take ~30 seconds of data using same labview vi. <u>IMPORTANT</u>: the static pressure inside the wind tunnel is *negative* (less than atmospheric); so, be sure that you record and code the correct sign.

(g) Measure the pressure distribution *below* stall:

i. Rotate the airfoil clockwise to achieve an angle of attack between 5–8 deg. Record the angle of attack.

ii. Start with the pressure tap located closest to the leading edge (white color tubing). Connect the tube to the same port on the transducer used for the static pressure reading previously. Check that the rest of the pressure taps on the airfoil are sealed.

iii. Take ~30 seconds of data. IMPORTANT: the static pressure on the top surface of the airfoil should be *negative* (less than atmospheric); therefore, be sure to record and code the correct sign.
iv. Repeat for the remaining eight pressure taps.
v. Rotate the airfoil counterclockwise to achieve a negative angle of attack (with the same absolute magnitude as that used above). This will correspond to the pressure distribution along the underside of the airfoil during a positive angle of attack. Repeat the static pressure measurements for all nine pressure taps. IMPORTANT: Be careful of the correct sign.

(h) Measure the pressure distribution *above* stall:

i. Rotate the airfoil clockwise to achieve a positive angle of attack between 14–16 deg.
ii. Repeat the pressure measurements using the same procedure as above.
iii. Rotate the airfoil counterclockwise to achieve a negative angle of attack (with the same absolute magnitude as that in the preceding steps).
iv. Repeat the pressure measurements.

4. Data analysis:

(a) Use (1) to obtain the Reynolds number based on chord length and approach flow speed. Use Bernoulli's equation to convert *dynamic* pressure obtained from the pitot-static probe to velocity (use the same equation as provided in previous labs).

(b) Use (2) to obtain the lift and drag coefficients based on the measured lift and drag forces. Be careful that your UNITS match.

(c) Use (3) to convert the measured static pressures along the airfoil to coefficients of pressure. Note, Ps on the data sheet represents p_∞ in (3).

Work Due

The work due for this laboratory will be in the format of a regular laboratory assignment with the following required parts.

- *Figures:*

1. Plot of C_L (y-axis) versus α (x-axis). The y-axis limits should be extend from 0 to your maximum lift coefficient value. The x-axis limits should extend from 0 to 18 deg.
2. Plot of C_D (y-axis) versus α (x-axis). Both axes should be plotted in linear coordinates (NOT logarithmic). The y-axis limits should be extend from 0 to your maximum drag coefficient value. The x-axis limits should extend from 0 to 18 deg.
3. Plot of C_L (y-axis) versus C_D (x-axis). This is often referred to as the "lift-drag polar". Plot your data using a marker (e.g., ○ or □). On the same plot, include three lines corresponding to previously published data at three different

Reynolds numbers (available for download from CANVAS). Use different linestyles to distinguish these three lines. Include a legend.

4. Plot of C_p (y-axis) versus x/c (x-axis) for an angle of attack *below* stall. In addition to the Reynolds number, also indicate the angle of attack in the title. Use different markers to distinguish the top and bottom airfoil surfaces. Include a legend.
5. Plot of C_p (y-axis) versus x/c (x-axis) for an angle of attack *above* stall. In addition to the Reynolds number, also indicate the angle of attack in the title. Use different markers to distinguish the top and bottom airfoil surfaces. Include a legend.

- *Matlab Code(s)* Attach all of your Matlab codes used to analyze the data and generate the required plots. In order to receive full credit, the code MUST be adequately commented.
- *Short Answer Questions:* Write 2–3 sentences in response to each question.
  1. Describe what happens to the lift coefficient before and after stall.
  2. Using appropriate fluids terminology, describe why the lift and drag increase with angle of attack up to the stall angle.
  3. Identify sources of uncertainty in the measurements and speculate how it could lead to discrepancies between the experimental results and published/accepted data.