TFES Lab (ME EN 4650)

Airfoil Aerodynamics

Textbook Reference: Section 9.7 (pp. 445–459) and Section 9.8 (pp. 459–474) from Pritchard, 8^{th} ed.

Objectives

(i) measure the lift and drag forces on a model NACA 0012 airfoil as a function of angle of attack using a lift/drag balance,

(ii) measure the static pressure along the top and bottom surfaces of the airfoil at two different angles of attack before and after stall, and

(iii) compare the measured data with published results of the lift, drag, and pressure coefficients at a similar Reynolds number.

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. An "airfoil" is the cross-sectional segment of a wing, as shown in Figure 1, that produces a net lift force under appropriate conditions. The first airfoil shapes investigated by NACA were designated by a four-digit number that represented the geometry of the airfoil. The different shapes are shown in the Appendix. After the mid 1930's, airfoil geometries became more sophisticated (as did the

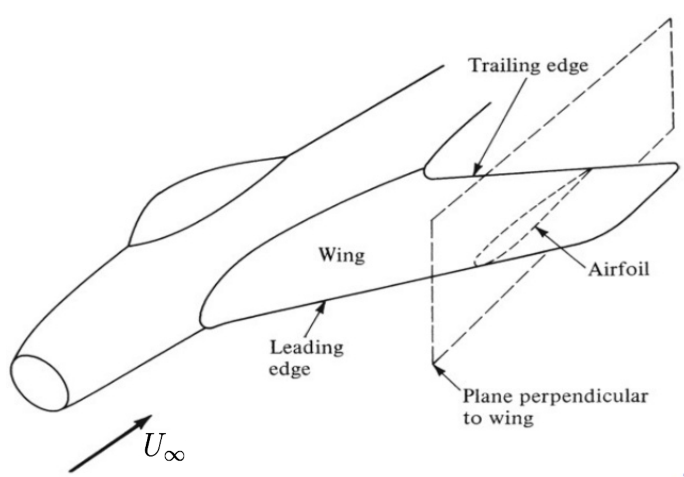


Figure 1. Schematic of an aircraft illustrating the definition of an "airfoil".

numbering scheme) in order to optimize lift while simultaneously decreasing drag. These airfoils were designed for high speed flow (i.e., high Reynolds number) as applicable to aircraft flight. Many of the initial four-digit NACA airfoils, such as the NACA 0015, are still used in the wind turbine industry today.

Airfoil Geometry

In the present experiment, we consider the aerodynamic performance of a NACA 0012 airfoil, shown in Figure 2. Table 1 lists the relevant terminology for airfoils. The lift force (F_L) acts perpendicular to the direction of the approach flow denoted by U_∞; 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 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 2) 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 2, 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.

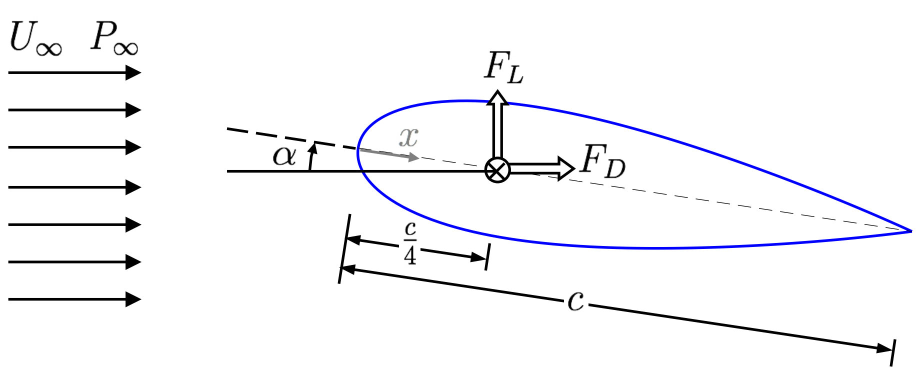


Figure 2. Schematic of the NACA 0012 airfoil at an angle of attack α. The net lift and drag forces act at the center of pressure, which for this airfoil occurs at a distance $x = c/4$ from the leading edge.

Table 1. Terminology relevant to airfoil aerodynamics.

Term	Symbol	Description
Freestream speed	U_∞	Far upstream speed of the wind relative to the airfoil speed
Freestream pressure	P_∞	Static pressure of the wind far upstream of the airfoil
Lift force	F_L	Component of the net pressure force acting perpendicular to the approach flow
Drag force	F_D	Component of the net pressure force acting parallel to the approach flow
Chord	c	Distance between leading and trailing edge along the chord line
Angle of attack	α	Angle between the chord line and the approach flow direction
Max thickness	t	Maximum distance between upper and lower surfaces of the airfoil
Span	s	Dimension of the airfoil across the flow
Planform area	A_p	Airfoil surface area projected onto a plane parallel with the approach flow
Aspect ratio	AR	Ratio of the chord over the span

Aerodynamic Performance

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

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

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 surface as referenced to the leading edge. With airfoils, a laminar boundary layer usually forms near the leading edge, grows for some short distance along the upper/lower surfaces, and then transitions to a turbulent boundary layer, before finally separating at some point prior to reaching the trailing edge of the airfoil, as depicted in Figure 3.

The coefficients of lift (C_L) and drag (C_D) for airfoils are defined as

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

The quantity $(1/2 \, \rho \, U_\infty^2)$ denotes the *dynamic* pressure associated with the freestream flow. In the case of the NACA 0012 airfoil section used in the present experiment, the planform area is given by $A_p = s \cdot c$. Note, the drag coefficient for an airfoil is NOT defined based

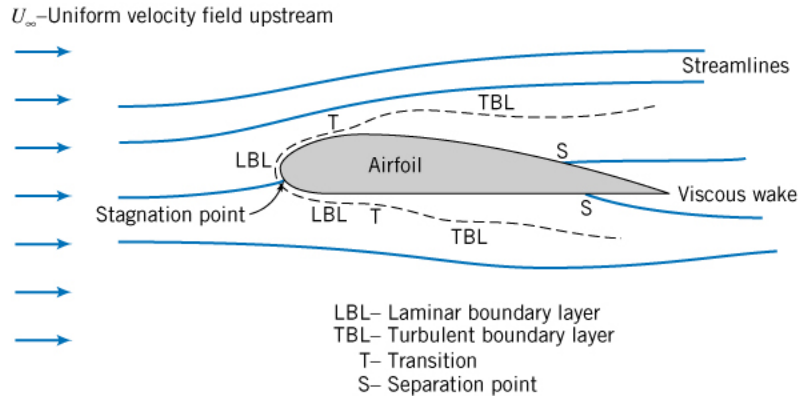


Figure 3. Illustration of the flow phenomena for an airfoil near zero angle of attack.

on the frontal area, but rather the planform area. The reason for this is because usually with airfoils, we are more concerned about its lifting capability. The lifting surface area is characterized by the planform area (A_p); hence, A_p is the appropriate characteristic area to use in the lift coefficient. In fact, most airfoils exhibit a maximum lift coefficient near 1, indicating that the quantity $(1/2\,\rho\,U_\infty^2\,A_p)$ is the appropriate characteristic force. For consistency then, and to allow for easy comparison of the lift coefficient magnitude, the drag coefficient for an airfoil is also defined using the same characteristic force as that for the lift coefficient.

The coefficient of pressure (C_p) is defined as

$$C_P = \frac{P_x - P_\infty}{1/2\,\rho\,U_\infty^2}, \tag{3}$$

where P_∞ denotes the *static* pressure of the freestream flow, and P_x is the *static* pressure along the airfoil surface and, hence, is a function of x. 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 toward the trailing edge. Laminar boundary layers also exhibit less skin friction. Therefore, to reduce drag, it is desirable to maintain a laminar boundary layer over as much of the airfoil surface as possible.

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 3) 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 upper surface of the airfoil, as shown by the flow visualization image in Figure 4. The term for this condition is *stall*. One of the objectives of this experiment is to identify the angle of attack at which stall occurs for the NACA 0012 airfoil at moderate Reynolds numbers ($Re_c \approx 1.5 \times 10^5$).

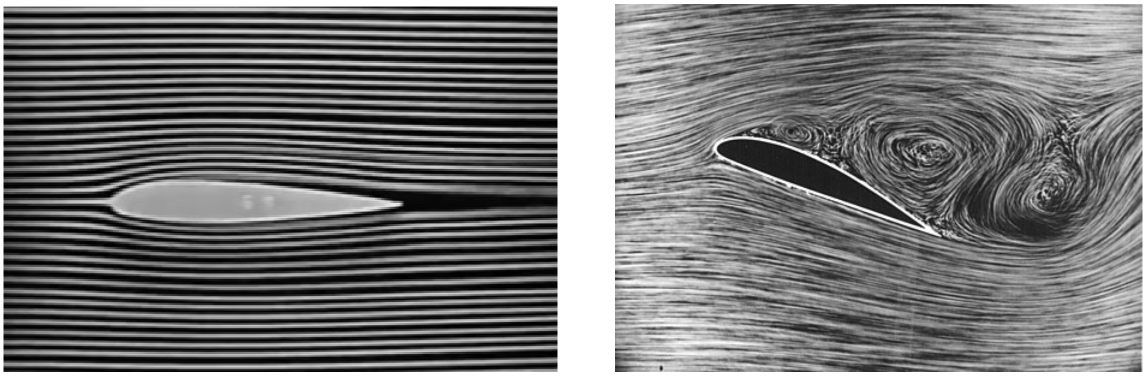


Figure 4. Flow visualization images, using the smoke line technique, of an airfoil at zero angle of attack (left) and at stall (right).

Experimental Setup

Experiments are conducted on a NACA 0012 airfoil in a wind tunnel with a 12 inch × 12 inch cross section. The chord length of the airfoil is $c = 4$ inches, the maximum thickness is $t = 0.48$ inches ($t = 0.12\,c$), 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, as shown in Figure 5: (a) one that mounts to the lift/drag balance to allow direct measurement of the lift and drag forces, and (b) one that has 9 small pressure taps along one surface of the airfoil to allow measurement of the static pressure distribution as a function of the axial distance along the chord, x/c. Both airfoils pivot thereby permitting a range of angles of attack to be investigated. The lift/drag airfoil pivots about a center support; while, the static pressure airfoil pivots about its end caps. The approach flow U_∞ may be varied by controlling the frequency of the power supplied to the wind tunnel fan.

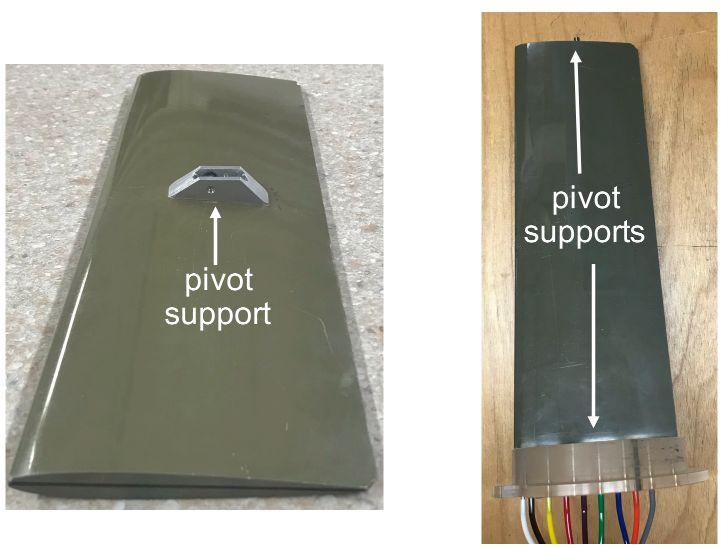


Figure 5. (left) Photograph of the airfoil used for the lift/drag measurements. (right) Photograph of the airfoil used for the static pressure measurements.

Table 2 lists the measurements that will be acquired in the experiment along with their native units. The lift/drag balance used to measure the lift and drag forces on the airfoil is comprised of two Linear Variable Displacement Transducers (LVDTs). The LVDTs measure the deflection of the thin support beams on the balance. A calibration is then utilized to convert deflection to applied load. Note, the native force unit on the balance is kg. Therefore, the lift and drag output must be multiplied by the gravitational constant to obtain the proper SI unit of N for force. Figure 6 shows a photograph of the lift/drag balance along with the instrumentation box. Importantly, NONE of the knobs on the instrumentation box should be rotated, except for the “pressure selection knob” in the lower-left corner. During operation, you will observe that the lift and drag forces tend to exhibit large fluctuations due to (i) noise in the LVDT electronics, and (ii) turbulence in the wake behind the airfoil. Therefore, the lift and drag output will be recorded with a data acquisition unit over a duration of 15 seconds at each angle of attack, and subsequently time-averaged during a post-processing step of the analysis.

Table 2. List of measurements acquired in the experiment with their native units. All pressure measurements utilize the same differential pressure transducer

Quantity	Symbol	Units	Instrument
Static pressure on airfoil	$P_x - P_{\text{atm}}$	in H_2O	surface taps on airfoil
lift + drag forces	F_L, F_D	kg	Lift/drag balance
Freestream velocity	$\frac{1}{2}\rho U_\infty^2$	in H_2O	Pitot-static probe

Figure 6. (left) Photograph of the lift/drag balance showing the location of the adjustment thumbwheels used to zero the lift and drag output. (right) Photograph of the front face of the instrumentation box used to measure lift, drag, and differential pressures.

The pressure measurements will be acquired with a *differential* pressure transducer that operates in a similar manner to the one used in the Circular Cylinder lab. In all cases, the pressure output represents the difference in pressures between the two ports on the back of the instrumentation box. The two ports are labeled $\mathsf{P}_{\mathsf{static}}$ and $\mathsf{P}_{\mathsf{total}}$. The pressure output $\mathsf{P}_{\mathsf{output}}$ displayed on the LCD screen on the front of the instrumentation box is

$$\mathsf{P}_{\mathsf{output}} = \mathsf{P}_{\mathsf{total}} - \mathsf{P}_{\mathsf{static}} \,. \tag{4}$$

The following explains how to configure the unit to obtain the pressure measurements.

→ <u>Freestream Velocity</u>: Set the "pressure select knob" to channel 0; move the Pitot-static probe into the freestream; and, connect the total- and static-pressure tubes of the Pitot-static probe to the ports on the back of the instrument box labeled TOTAL and STATIC, respectively. In this configuration, the pressure displayed on the LCD screen is

$$P_{\mathsf{output}} = \tfrac{1}{2}\,\rho\, U_\infty^2 \,. \tag{5}$$

→ <u>Airfoil Static Pressure</u>: Set the "pressure select knob" to any channel between 1 through 9. In this configuration, the pressure readout on the PRESSURE display is

$$P_{\mathsf{output}} = P_x - P_\infty \,, \tag{6}$$

where P_x denotes the static pressure on the airfoil surface corresponding to a location of x along the chord. The x-coordinates of the nine pressure taps are listed in Table 3.

Table 3. Location of pressure taps on the airfoil.

pressure channel	x (in)	x/c
1	0.2	0.05
2	0.4	0.1
3	0.8	0.2
4	1.2	0.3
5	1.6	0.4
6	2.0	0.5
7	2.4	0.6
8	2.8	0.7
9	3.2	0.8

Laboratory Procedure

Before starting, make a note that none of the SCAN knobs on the front of the instrumentation box (see Figure 6) should be adjusted, as this will invalidate the preset calibrations.

1. Air Properties
   (a) Measure the room temperature using the thermometer.
   (b) Measure atmospheric pressure using the barometer.
   (c) Use the ideal gas law to calculate the local air density.
2. Lift and Drag Experiment
   (a) Start the data acquisition program: `AirfoilLab_LiftDrag-Fixed Time.vi`.
   (b) Mount the lift/drag balance to the bottom of the wind tunnel such that the streamlined sting extends into the test section, with the rounded portion of the streamlined sting facing upstream, as shown in Figure 7.

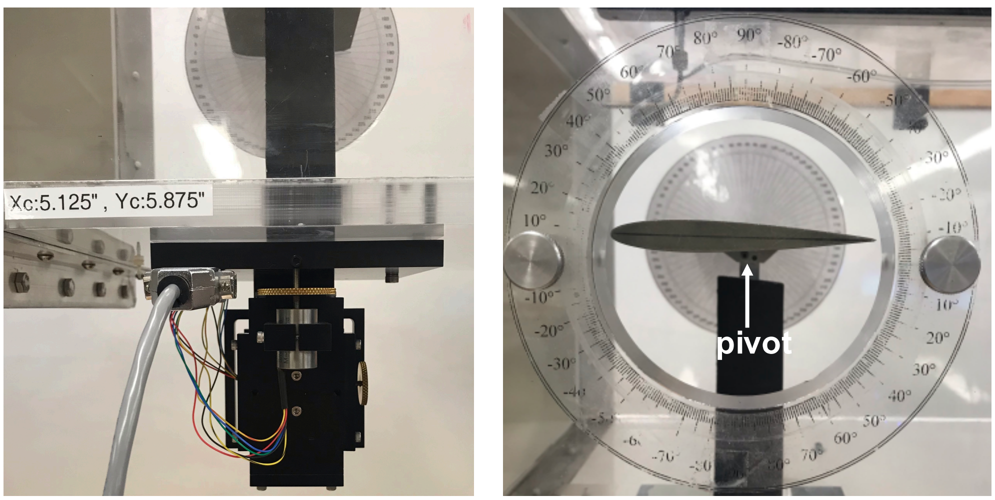


Figure 7. (left) Photograph of the lift/drag balance mounted underneath the wind tunnel. (right) Photograph of the lift/drag airfoil mounted inside the wind tunnel.

   (c) Verify that the cable from the lift/drag balance is connected to the port labeled "Dynamometer" on the back of the instrument box.
   (d) Secure the airfoil to the mounting point at the top of the sting of the lift/drag balance by sliding the pivot (located on the underside of the airfoil) over the top of the sting. A cap screw is provided to fasten the airfoil pivot to the sting. Use a long T-handle 3/32″ hex wrench to tighten the cap screw. DO NOT OVERTIGHTEN the cap screw!
   (e) Set the angle of attack of the airfoil to $\alpha = 0$ deg. Use the protractor taped to the far sidewall of the wind tunnel as a guide to manually set the angle. Use the set screw to GENTLY tighten the airfoil in this configuration.

(f) Zero the lift/drag balance so that the readout on the data acquisition program displays 0.00 for both the lift and drag.

Importantly, the display on the front of the instrument box for the lift and drag will NOT exactly match the display in the data acquisition program. The lift and drag instrumentation was pre-calibrated to display the actual force in units of kg on the data acquisition program. However, you still need to "zero" the display to remove the DC Offset.

i. Turn the brass thumbwheel for "drag", located on the lift/drag balance, until the display labeled "Drag" on the instrument box reads "0.00" kg. NOTE, the thumbwheel for "drag" is the smaller of the two thumbwheels; see Figure 6.

ii. Turn the brass thumbwheel for "lift", located on the lift/drag balance, until the display labeled "Lift" on the instrument box reads "0.00" kg. NOTE, the thumbwheel for "lift is the larger of the two thumbwheels; see Figure 6.

(g) Zero the pressure transducer reading on the front LCD display of the instrument box. To do this, insert a #1 flathead screwdriver into the hole labeled Pressure Zero on the top of the instrument box. Carefully, maneuver the screwdriver until you can feel it engage with the potentiometer screw inside the instrument box. GENTLY turn the screw (it is very sensitive) until the LCD display reads 0.00.

(h) Take a baseline reading from the dynamometer at no flow. Start sampling data from the lift/drag balance by pressing the white arrow at the upper lefthand corner of the toolbar in the data acquisition window on the computer.

Note, you will average the data in the lift and drag columns of this file in order to get the DC Offset of the instrument. You will then subtract this value from all subsequent readings.

(i) Set the fan frequency to a setting between 40–50 Hz in order to achieve the desired chord Reynolds number, Re_c, between 1×10^5 and 1×10^6.

(j) Measure the freestream flow speed, U_∞:

i. Position the Pitot-static probe near the test section entrance, so that the X AXIS POSITION reads 1.5 inch, and the Y AXIS POSITION reads 9.5 inch on the front of the instrument box. In this position, the Pitot-static probe will be slightly upstream and above the airfoil.

ii. Connect the flexible tubing from the Pitot-static probe to the TOTAL and STATIC pressure ports on the BACK of the instrument box, as shown in Figure 8.

iii. Set the "pressure select knob" on the front of the instrumentation box to channel 0; see Figure 8.

iv. Observe the differential pressure value on the PRESSURE display of the instrumentation box and record it on your data sheet.

v. Leave the Pitot-static probe at this location for of all experimental measurements.

(k) Record lift/drag data at the current angle of attack, F_L and F_D:

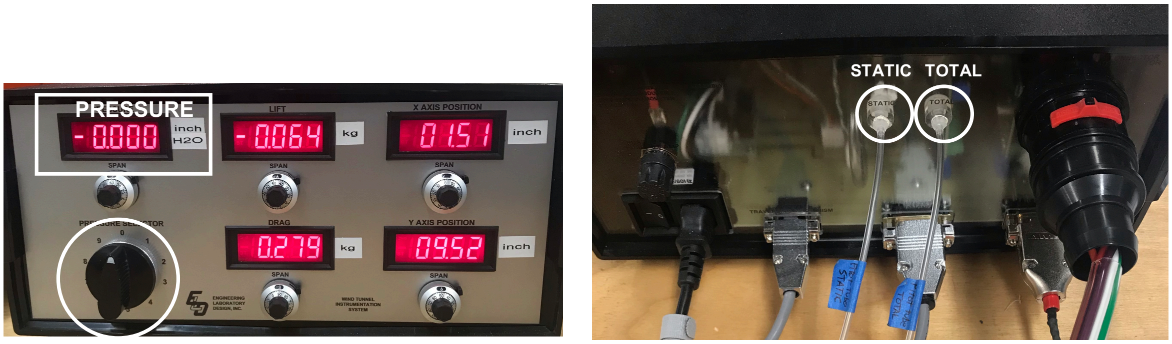


Figure 8. (left) Photograph of the front of the instrumentation box showing the location of the PRESSURE display and "pressure select knob", highlighted in white. Importantly, none of the SCAN knobs on the front of the instrumentation box should be adjusted. (right) Photograph of the back of the instrumentation box showing the location of the STATIC and TOTAL pressure ports, also highlighted in white.

i. Enter a filename on the data acquisition program in the appropriate textbox. Write this filename on your data sheet.

ii. Make sure the sampling time is set to 15 sec.

iii. Press the white arrow on the toolbar at the top of the window to start recording data.

(l) Adjust the angle of attack in 1° increments (or as close to 1° as possible) and repeat the previous step for angles of attack between 1°–16°. Note, the data acquisition program will automatically overwrite the existing data file, if you do not change the file name. You may find it helpful to lower the wind tunnel fan speed while adjusting the angle of attack. Be sure to set the wind tunnel speed back to the same setting, and allow a few seconds for the speed to become steady, before recording your data

3. Airfoil Static Pressure Distribution

(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 CAREFULLY place off to the side of the instrument box.

(c) Seal the opening in the bottom of the wind tunnel with the provided acrylic plate.

(d) Slide the airfoil with the surface pressure taps through the side access hole and position it horizontally across the test section of the wind tunnel. Be sure the post on the end of the airfoil mates securely with the hole in the farside of the wind tunnel side wall. Also ensure the airfoil is oriented such that the pressure taps are located on the top surface of the airfoil.

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

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

i. Ensure that the tubing from the *static*-side of the Pitot-static probe is still connected to the port labeled STATIC on the back of the instrumentation box. Note, all pressure channels on the "pressure select knob" are measured relative to the pressure on the STATIC port.

ii. Rotate the airfoil clockwise to achieve an angle of attack near $+5°$. Read off the angle of attack using the provided protractor and record it on your data sheet.

iii. Turn the "pressure select knob" dial on the front of the instrument box to 1. The readout on the PRESSURE display will indicate the gauge pressure at the first tap closest to the leading edge of the airfoil.

iv. Record the pressure value on your data sheet in the column labeled "Top Surface". Note, you will need to visually average the values shown on the display.

v. Turn the "pressure select knob" dial to the next setting and record the reading on the PRESSURE display. Repeat for the remaining pressure taps.

vi. 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 and record on your data sheet in the column labeled "Bottom Surface".

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

i. Rotate the airfoil clockwise to achieve a positive angle of attack near $+12°$.

ii. Repeat the pressure measurements using the same procedure as above and record your measurements on your data sheet. Note, because the flow is completely separated from the airfoil after stall, the static pressure measurements tend to fluctuate much more in this position. You will need to visually average the values shown on the display. Recording the pressure values to within two significant digits is sufficient.

Data Analysis

1. Lift and Drag Coefficients

(a) Time-Average Data

The first task in the data analysis is to time average the data acquired in each data file, to obtain the mean and standard deviation. Note, the first column in the data file represents the "lift" and the second column represents the "drag". The units of the numbers in the data file are kg.

The lift and drag measurements will tend to exhibit large fluctuations in time, due to (i) vibration of the airfoil since it is supported by a single pivot point, as well as (ii) electronic noise in the LVDT signal conditioner. Figure 9 illustrates a sample time series of the lift force, showing the mean ($\overline{F}_L$) and standard deviation (σ_{F_L}) of the signal. The sampling rate of the signal is 50 Hz (i.e., 50 samples are digitized every second). If you zoom in, you can notice that the signal looks

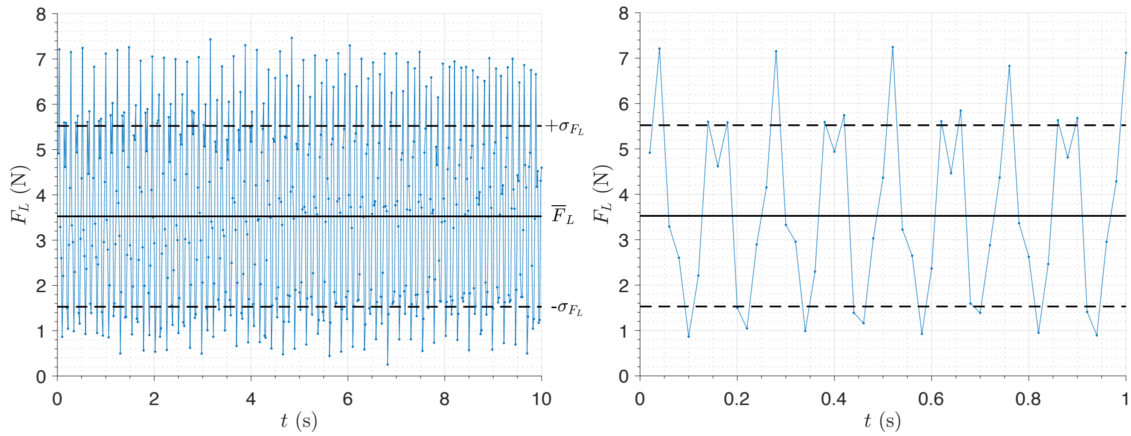


Figure 9. Plot of the lift force as a function of time as measured from the LVDT sensor. (left) Entire record. (right) Zoomed-in region showing the sinusoidal nature of the signal. The solid black line represents the mean value and the dashed black lines represent $\pm$ one standard deviation.

sinusoidal with a well-defined characteristic frequency of about 8 Hz. This is likely the excitation frequency of the magnetic coil in the LVDT sensor, as this frequency appears in all of the output signals regardless of angle of attack.

(b) Calculate Uncertainty

The second task is to calculate the uncertainty in the mean values using the standard error of the mean. Then, the lift and drag at each angle of attack can be specified as a range,

$$F_L = \overline{F}_L \pm 2\,\sigma_{\overline{F}_L} \qquad \text{and} \qquad F_D = \overline{F}_D \pm 2\,\sigma_{\overline{F}_D}\,. \tag{7}$$

In the above expressions, the factor of 2 is used so that we can claim a 95% confidence interval in the specified range on the mean value. This signifies there is only a 5% chance that the true mean is outside of the specified range. The uncertainty is given by

$$\sigma_{\overline{F}_L} = \frac{\sigma_{F_L}}{\sqrt{N}} \qquad \text{and} \qquad \sigma_{\overline{F}_D} = \frac{\sigma_{F_D}}{\sqrt{N}}\,, \tag{8}$$

where N represents the number of <u>independent</u> samples in the series. Because of the sinusoidal nature of the noise (with a frequency of about 8 Hz), there are only 8 independent data points per second. If you record for 15 seconds, then $N = 15 \cdot 8 = 120$.

To obtain the uncertainty in the lift and drag <u>coefficients</u>, we can employ a first-order Taylor's series expansion of the expressions given in (2). This leads to the following

$$C_L = \frac{\overline{F}_L}{\frac{1}{2}\,\rho\, U_\infty^2\, A_p} \pm 2\left(\frac{\sigma_{\overline{F}_L}}{\frac{1}{2}\,\rho\, U_\infty^2\, A_p}\right) \quad \text{and} \quad C_D = \frac{\overline{F}_D}{\frac{1}{2}\,\rho\, U_\infty^2\, A_p} \pm 2\left(\frac{\sigma_{\overline{F}_D}}{\frac{1}{2}\,\rho\, U_\infty^2\, A_p}\right) \tag{9}$$

(c) Comment on Observed Bias

The lift/drag balance used in the present experiment has a noticeable bias (also referred to as systematic error). Because the NACA 0012 airfoil is symmetric, there should be no lift force generated at zero angle of attack ($\alpha = 0°$). However, you will notice this is not the case for your data, as illustrated in Figure 10. The reason for the bias in the measurements is believed to be due to the fact that the pivot support on the airfoil is located at the center of mass ($x = \frac{c}{2}$), rather than the center of pressure ($x = \frac{c}{4}$). This means there is a net moment about the pivot point that creates induced forces on the lift/drag balance. A better design would be to place the pivot point at $x = \frac{c}{4}$ to avoid contamination of the aerodynamic lift and drag by these induced forces.

Note, your drag coefficient results will be similarly contaminated with bias. We can notice this by comparing the C_D value at $\alpha = 0°$ with previously published results (R.E. Sheldahl and P. C. Klimas, "Aerodynamic characteristics of seven symmetrical airfoil sections through 180-degree angle of attack for use in the aerodynamic analysis of vertical axis wind turbines", Sandia National Laboratories Report, SAND80-2114, 1981) as shown in Figure 10. These results indicate that, for the NACA 0012 airfoil, the drag at zero angle of attack (C_{D_0}) decreases with chord Reynolds number according to the following empirical relation

$$C_{D_0} = \begin{cases} Re_c^{-0.383} & \text{for} \quad Re_c \leq 6.2 \times 10^5\,, \\ 0.00645 & \text{for} \quad Re_c > 6.2 \times 10^5. \end{cases} \tag{10}$$

The data you collect in the lab will tend to exhibit a drag coefficient at zero angle of attack that is slightly more than twice the value of that based on (10).

In terms of professional ethics, dealing with bias can be complicated. One can

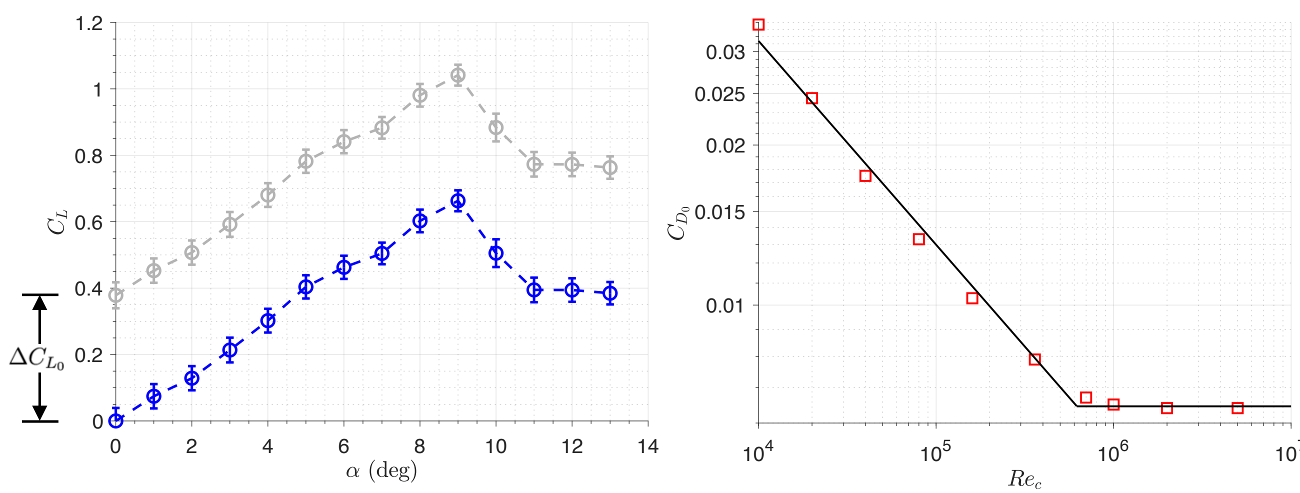


Figure 10. (left) Lift coefficient versus angle of attack for the NACA 0012 airfoil at $Re_c = 1.5 \times 10^5$. ○ Raw data. ○ Data compensated for bias by uniformly shifting all data an amount ΔC_{L_0} such that $C_L = 0$ at $\alpha = 0°$. (right) Drag coefficient at $\alpha = 0°$ versus chord Reynolds number for the NACA 0012 airfoil. □ Published experimental data. — Curve fit.

attempt to correct for bias; but in doing so, you need to be very clear in your technical communication about the operations that have been performed on the data. For example, one could attempt to compensate for the observed bias in the present data by shifting the C_L and C_D data a constant amount, such that $C_L(\alpha = 0°) = 0$ and $C_D(\alpha = 0°)$ matches the published data. However, this will cause the maximum C_L value to be underestimated, because the bias error does not appear to be uniform with angle of attack. In addition, this correction for C_D is not enough to compensate for the large over-estimation of drag after stall. In general, it is best not to attempt to correct for bias in the data, unless you have direct evidence or knowlege about the magnitude of your bias error. Instead, it is preferable to redesign the experimental setup to significantly reduce or eliminate bias error altogether.

2. Coefficient of Pressure

   From the static pressure measurements along the top and bottom surfaces of the airfoil, you need to calculate the coefficient of pressure, C_P, based on (3). Note, in the lab, you already measured the relative static pressure, $P_x - P_\infty$. Therefore, you only need to divide the measurements by the *dynamic pressure of the freestream* in order to calculate C_P. Be sure to verify that your units are consistent.

Figures

Captions

A meaningful and comprehensive caption must accompany all figures. For the three figures, the caption is placed *below* the figure and includes the label Figure 1x., where x denotes the letter a–c according to the plot order listed below. Note, because plot 1c includes two subplots side-by-side, the caption needs to differentiate the two subplots using "(left)" and "(right)" labels — see Figure 4 of this Handout as an example.

Plots

1a. Plot of C_L (y-axis) versus α (x-axis) comparing your data to previously published results. Download the provided Matlab figure file entitled: `NACA0012_CL.fig`, open this file in Matlab, and plot your data on this figure. Include error bars on C_L that represent the uncertainty to within a 95% confidence interval. The y-axis limits should extend from 0 to 1.4. The x-axis limits should extend from 0 to 20 deg. Plot your data using a marker (e.g., ○ or □) and connect the markers with a dashed line to better discern the trend. Include the chord Reynolds number of your data in the legend.

1b. Plot of C_D (y-axis) versus α (x-axis) comparing your data to previously published results. Download the provided Matlab figure file entitled: `NACA0012_CL.fig`, open this figure in Matlab, and plot your data on this figure. Include error bars on C_D that represent the uncertainty to within a 95% confidence interval. The y-axis limits should be extend from 0 to 0.3. The x-axis limits should extend from 0 to 20 deg. Plot your data using a marker (e.g., ○ or □) and connect the markers with a dashed line to better discern the trend. Include the chord Reynolds number of your data in the legend.

1c. Create two suplots of the pressure coefficient C_p (y-axis) versus distance along the chord x/c (x-axis). The two subplots should be placed side-by-side. The **left** subplot should contain the data for an angle of attack *below* stall; while the **right** subplot should contain the data for an angle of attack *above* stall. In both subplots, use ○ for the top surface and □ for the bottom surface. Include a legend in both subplots. Include a single figure caption for the two subplots. In the figure caption, state the chord Reynolds number and angles of attack for both subplots, as well as the actual stall angle.

Short-Answer Questions

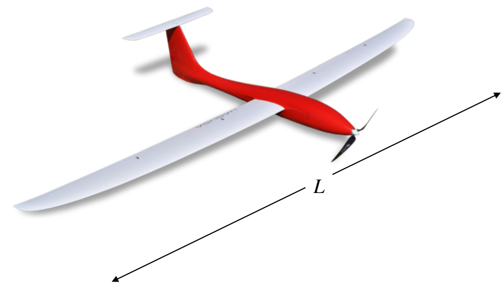
2a. State the angle of attack at which stall occurs (α_{stall}) for your data. Your response should in the following form (where XX denotes the value from your data): "Based on the present lift and drag measurements at Re_c =XX, stall is observed to occur at an angle of attack of about XX°." Describe how your measured lift and drag coefficients vary with angle of attack. [3–4 sentences]

2b. State the average percent uncertainty in your C_L and C_D measurements to within a 95% confidence interval. Describe how the uncertainty in your experimental measurements varies with angle of attack, if at all. [2–3 sentences]

2c. Compare your data to previously published results and examine Re_c trends:

- State 2 ways each that your measured C_D and C_L values disagree with previously published results at a similar Re_c. Be specific in your response. For example, do not simply state that your values are larger than the published data, but provide the actual numeric values, both for your data and the previously published results. [4 sentences]
- Examine the trends in the previously published results, and state 2 ways each that C_D and C_L vary with Re_c. [2–4 sentences]

2d. Imagine that you are trying to use your experimental results to select the correct size electric motor to power a small Unmanned Aerial Vehicle (UAV) such as the one shown, where the wings are comprised of NACA 0012 airfoils. The total wingspan is $L = 4$ ft. and the chord is $c = 6$ in. Assume that the cross-section of each wing is uniform, i.e., the wings do not taper and the planform area looks rectangular. State the speed V_w at which the UAV would have to be flying in order for your wind tunnel results to be applicable. If the UAV were cruising at V_w, state the maximum mass it could have (to ensure that it would remain flying at this speed). Use an air density and air viscosity consistent with what is typically observed in Salt Lake City at around 23°C. [2–3 sentences with equations]



APPENDIX I: NACA Airfoil Shapes

The National Advisory Committee for Aeronautics (NACA) was a U.S. federal agency founded in 1915 under president Woodrow Wilson to promote aeronautical research, primarily for military purposes in the midst of World War I. NACA designed and tested a multitude of different airfoil shapes in the hopes of finding the best ones for military and civilian aviation. To do this, NACA built seven different large-scale wind tunnels, over the span of a decade (1920–30), that allowed for the aerodynamic testing of both airfoils and scaled-down aircraft. Figure 11 illustrates the different types of airfoil shapes that were designed and tested.

An "airfoil" is the cross-sectional shape of a wing, blade, or sail. Use of the word "airfoil" to describe such a shape implies that it is capable of generating a lift force under appropriate conditions. The geometry of a typical airfoil is illustrated in Figure 12. The chord line of an airfoil is the straight line that connects the leading and trailing edges of the airfoil. The length of the chord line is denoted as c. The maximum thickness of the airfoil is denoted as t. The coordinates of the airfoil are given by (x, y), where x represents the coordinate along the chord line and y is the coordinate normal to the chord line. The coordinate $(x, y) = (0, 0)$ denotes the leading edge of the airfoil. The camber line (also referred to as the "mean line") represents the curve that is halfway between the upper and lower surfaces of the airfoil, and

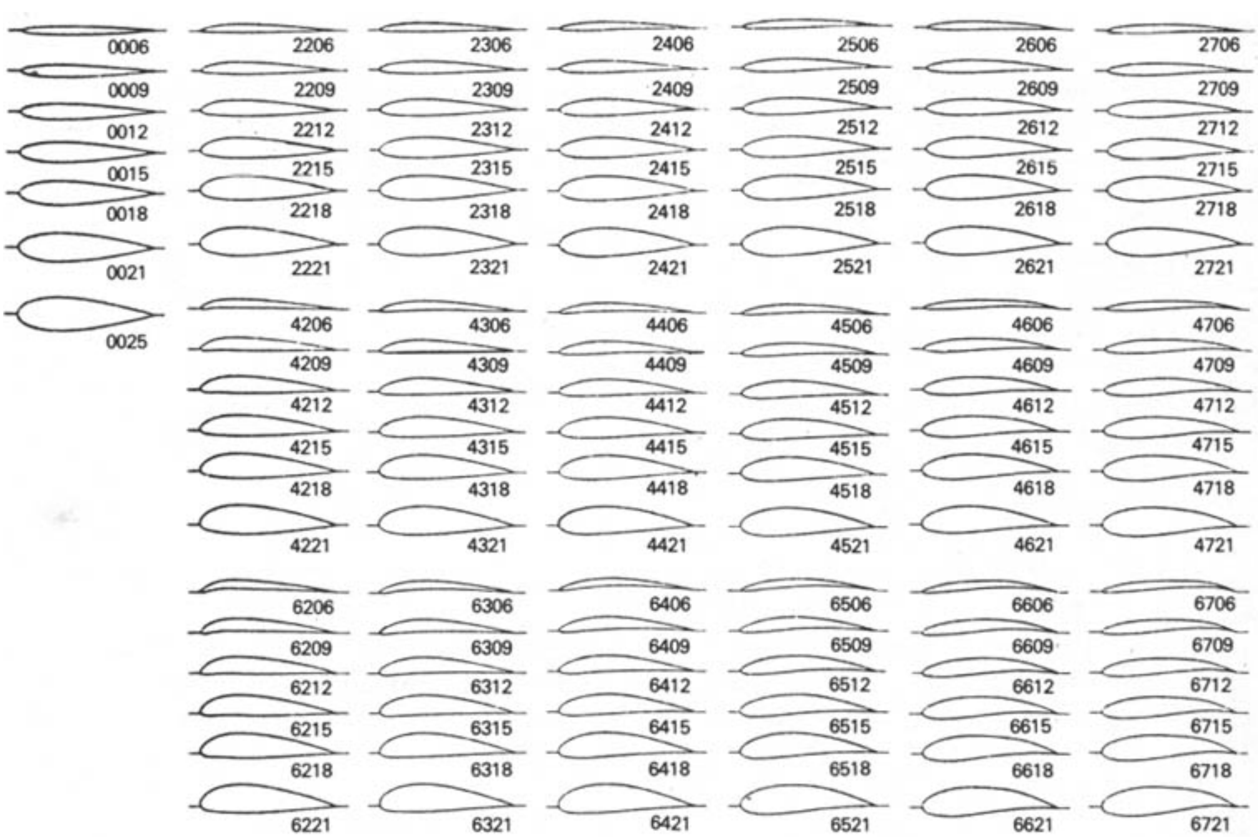


Figure 11. Chart comparing the various airfoil shapes tested by NACA from 1920–1958.

thus splits the airfoil into two halves with the same thickness as a function of x. A symmetric airfoil is one in which the chord line and camber line are coincident.

Four-Digit Symmetric Airfoils

The first types of airfoils investigated by NACA were symmetric airfoils and were assigned a number NACA 00XX, where XX is the percent maximum thickness relative to the chord. For example, with the NACA 0012 airfoil, $t = 0.12\,c$. The profile shape of the four-digit symmetric NACA airfoils (x, y_t) is determined from the following equation

$$\frac{y_t}{c} = \pm 5\,t\left[0.2969\left(\frac{x}{c}\right)^{1/2} - 0.1260\left(\frac{x}{c}\right) - 0.3516\left(\frac{x}{c}\right)^2 + 0.2843\left(\frac{x}{c}\right)^3 - 0.1015\left(\frac{x}{c}\right)^4\right]. \quad (11)$$

The maximum thickness of these airfoils, by default, occurs at $x/c = 0.3$. The leading edge of these airfoils is described by a circular arc with a radius of curvature equal to

$$r_0 = 1.10\,t^2\,. \quad (12)$$

Note, because the NACA 00XX airfoils are symmetric, they do not generate a net lift force at zero angle of attack.

Four-Digit Cambered Airfoils

NACA designed airfoils with a camber in order to achieve higher lift coefficients. This allowed aircraft to fly at slower speeds and higher angles of attack before stalling, which is critical during landing, for example. The numbering scheme for these airfoils is mpXX, where 100 m is the maximum percent camber relative to c, and 10 p is the location of the maximum camber relative to c. For example, in the NACA 2412 airfoil, m=2%, p=40%, and $t/c = 12$. The equation describing the mean camber line is

$$y_c = \begin{cases} \frac{m}{p^2}\left(2\,p\left(\frac{x}{c}\right) - \left(\frac{x}{c}\right)^2\right), & 0 \le x \le p\,c\,, \\ \frac{m}{(1-p)^2}\left((1-2\,p) + 2\,p\left(\frac{x}{c}\right) - \left(\frac{x}{c}\right)^2\right), & p\,c \le x \le c\,, \end{cases} \quad (13)$$

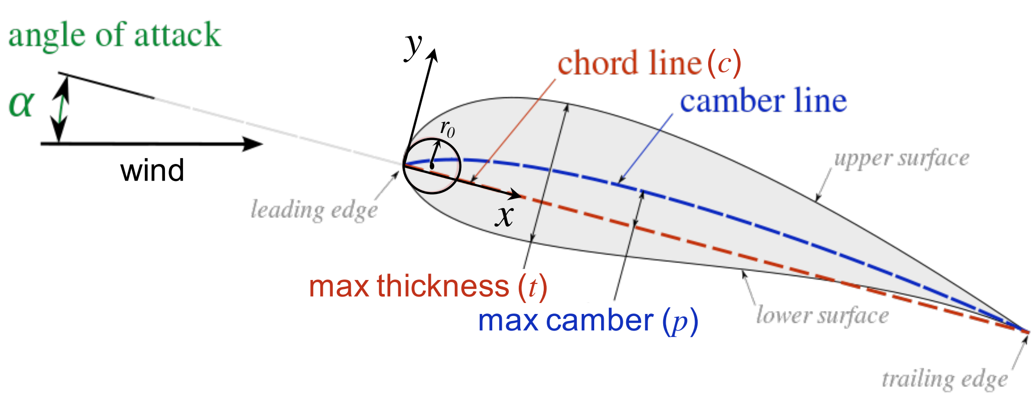


Figure 12. Cambered airfoil geometry.

where y_c denotes the y-coordinate of the mean camber line. The coordinate for the upper surface (x_U, y_U) and lower surface (x_L, y_L) of the airfoil are given by

$$x_U = x - y_t \sin\theta, \tag{14}$$
$$x_L = x + y_t \sin\theta, \tag{15}$$
$$y_U = y_c + y_t \cos\theta, \tag{16}$$
$$y_L = y_c - y_t \cos\theta, \tag{17}$$

where y_t is the corresponding y-coordinate for the equivalent symmetric airfoil given in (11), and

$$\theta = \tan^{-1}\left(\frac{dy_c}{dx}\right). \tag{18}$$

The derivative terms appearing in the equation for θ is

$$\frac{dy_c}{dx} = \begin{cases} \frac{2\,m}{p^2}\left(p - \frac{x}{c}\right), & 0 \le x \le p\,c, \\ \frac{2\,m}{(1-p)^2}\left(p - \frac{x}{c}\right), & p\,c \le x \le c\,. \end{cases} \tag{19}$$

Five-Digit and Higher Airfoils

NACA went on to develop airfoils with more complicated shapes, including those in their five-digit series, 1- and 16-series, 6-series, 7-series, and 8-series. The four-digit, five-digit, and 6-series airfoils have enjoyed use in many engineering applications, such as general aviation, helicopter blades, turbine blades, missile/rocket fins, ship propellers, and supersonic jets. Starting with the 1-series, NACA began designing airfoil shapes based on airfoil theory, rather than strictly geometric relationships that were used prior. The 6-series airfoils were based on improved theoretical methods to achieve a prescribed lift coefficient. The 7-series airfoils were designed to maximize laminar flow over the airfoil in order to reduce drag. However, these airfoils suffered from a reduced lift coefficient and poor stall behavior, and were seldom used in real applications. The 8-series airfoils were designed for flight at supercritical speeds, but also did not end up being used in any real-life applications. An excellent reference on the subject is F. Riegels, “Aerofoil Sections”, Butterworths (1961).