ASEN 2002
Aerodynamics Experimental Laboratory 2
Aerodynamics of a Cambered Airfoil

Assigned Wednesday 14 November

Lab Reports due Wednesday 12 December at 11:59PM

OBJECTIVES

- *Learn basic concepts and definitions associated with two-dimensional flow around airfoils.*
- *Gain an understanding of the origins of Lift and Drag through measuring and integrating the surface pressure distribution on a cambered airfoil.*
- *Observe how changes in the Lift and Drag force coefficients are related to the angle of attack (flow incidence angle) and the free-stream velocity.*
- *Develop an awareness of sources of error and error analysis.*

REQUIRED DELIVERABLES

- *Attendance at every lab period is required. Instructions for weekly tasks and the individual report will be presented during the scheduled lab time.*
- *Prepare a written brief report of the results of your laboratory exercises. Use the guidelines that will be provided to complete your report.*

TEXT REFERENCES

"Introduction to Flight" 8th ed. by Anderson

- ***Viscous Flow and Boundary Layers:*** *4.20, 4.21, 4.18*
- ***Airfoil Nomenclature and Data:*** *5.2, 5.4*
- ***Pressure and Force Coefficients:*** *5.3, 5.6*
- ***Obtaining Forces from Pressure:*** *5.7*

"Introduction to Error Analysis" 2nd ed. by Taylor

- ***Error and Uncertainty Analysis:*** *Chapters 1-4*

SUMMARY

Airfoils compose the fundamental cross-sectional geometry in an aircraft's wing and thus play a critical role in the production of lift and drag on aerodynamic vehicles. Furthermore we fundamentally understand that the pressure and shear-stress distributions on the physical surfaces of a body provide the source of all aerodynamic forces applied to that body. As a result, quantifying and integrating the pressure distribution around an airfoil geometry will provide you a fundamental understanding of the origins of lift and drag on aerodynamic vehicles.

To this end we have selected a canonical airfoil geometry, namely a cambered Clark Y-14 airfoil, which is instrumented with 19 flush mounted pressure taps around the surface. You will measure the pressure distribution on the airfoil surface for variations in angle of attack and free-stream airspeed. From this data you can then compute the resultant forces and determine the aerodynamic performance (i.e. lift and drag) of the wing section. This will fundamentally show where the aerodynamic forces originate, as well as how they depend on the flow conditions!

CLARK Y-14 CAMBERED AIRFOIL BACKGROUND

The Clark Y Airfoil is a commonly used wing section profile that can be found in many well-known aircraft. Specifically it is regularly used in model aircraft due to its relatively simple flat bottom construction. At a larger scale it can also be found in the *Spirit of St. Louis*, the plane flown by Charles Lindbergh in his first successful solo flight across the Atlantic Ocean. For our laboratory exercise we will use a modified version of the Clark Y airfoil configuration, the details of this configuration and the testing procedures are each outlined below.

Airfoil Components and Theory

The design of a wing generally begins with the airfoil. The airfoil schematic in Figure 1 shows some of the associated nomenclature. For more information you are referred to section 5.2 in the text by Anderson.

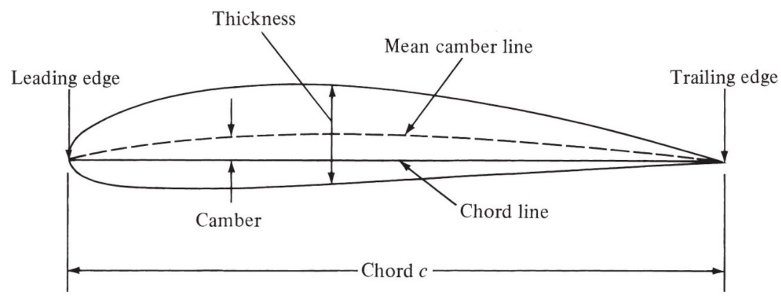

Figure 1: Airfoil schematic with relevant nomenclature.

As we have discussed in class, all aerodynamic forces are the result of pressure and shear stress distributions around the surface of a body. In this lab we will show that the lift force produced by an airfoil is primarily the result of its pressure distribution. Furthermore we will study how the pressure distribution and thus lift force scales with the airspeed and varies with the angle of attack (incidence) between an airfoil and the free stream velocity. In order to accurately make these comparisons it is common practice to write the pressure and force components in terms of dimensionless coefficients. This allows experimental measurements to be made on a small model in a wind tunnel and scaled up to the full-size configuration. It also allows us to understand the dependence of a measurement on the other experimental variables. As an example of an important dimensionless quantity we define the coefficient of pressure:

$$C_p = \frac{p - p_\infty}{\frac{1}{2}\rho_\infty V_\infty^2} = \frac{p - p_\infty}{q_\infty}$$

where the subscript ∞ denotes the free-stream value far upstream of an object in the undisturbed flow and q_∞ is the dynamic pressure. This coefficient of pressure (or pressure coefficient) contains information for how pressure varies on the surface of an immersed object. Furthermore, using Bernoulli's equation, we can directly relate the local speed at different points on the surface of the object to the coefficient of pressure.

$$C_p = 1 - \frac{V^2}{V_\infty^2}$$

In a similar manner we can define a coefficient of force, and will do this for both Lift and Drag Forces,

$$C_L = \frac{L}{\frac{1}{2}\rho_\infty V_\infty^2 S} = \frac{L}{q_\infty S}$$

$$C_D = \frac{D}{\frac{1}{2}\rho_\infty V_\infty^2 S} = \frac{D}{q_\infty S}$$

where L and D represent the lift and drag forces respectively, and S is a reference surface area. For a rectangular wing section we will simply define the reference area as the chord length, c, multiplied by the wing span, b,

$$S = c \cdot b.$$

The coefficients of lift and drag for an airfoil are commonly reported values in literature that define the performance of an airfoil. The coefficients of a Clark Y-14 airfoil were experimentally measured and reported in a NACA Technical Report, specifically number 628 from 1938. Figure 2 is taken from page 42 of this report and displays the variation in all of these coefficients versus angle of attack at a Reynolds Number of $Re = 3 \cdot 10^2$.

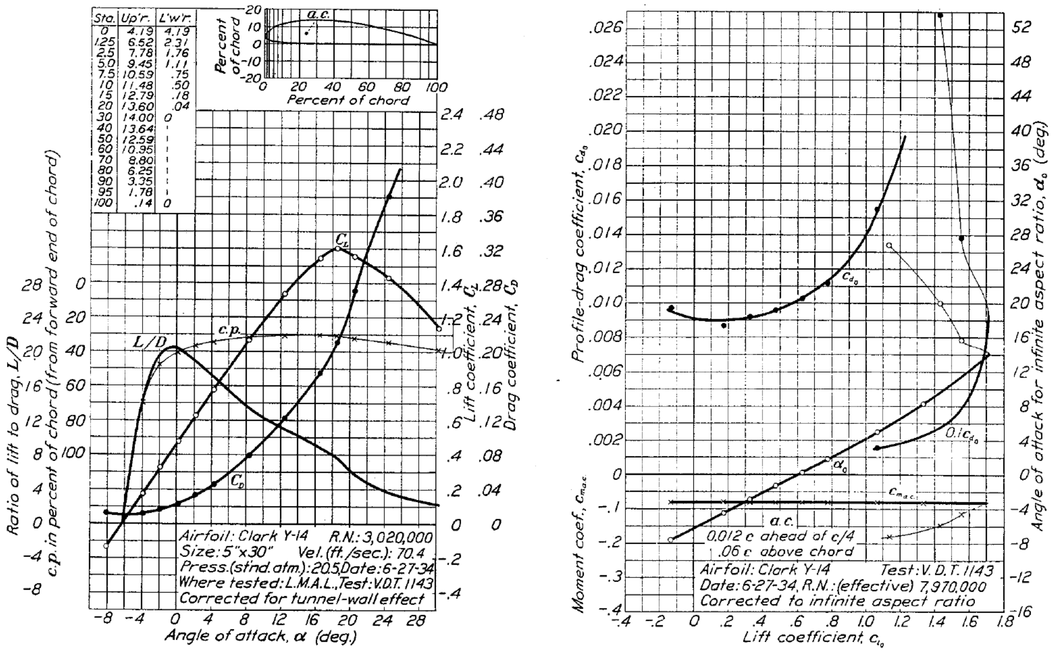

Figure 2: Aerodynamic coefficients for a Clark Y-14 airfoil from NACA Technical Report 628 (pg. 42).

We have just defined the coefficients of lift and drag (C_L and C_D) for a finite span wing where the wing area is defined as S. When we are working with airfoils (i.e. an infinite span wing), as we are in this laboratory assignment, we will typically define a sectional coefficient of lift and drag

$$C_l = \frac{l}{\frac{1}{2}\rho_\infty V_\infty^2 c} = \frac{l}{q_\infty c}$$

$$C_d = \frac{d}{\frac{1}{2}\rho_\infty V_\infty^2 c} = \frac{d}{q_\infty c}$$

where l and d are the lift and drag forces per unit span of the wing surface. Note that these two coefficients will be equal for a rectangular wing that has a uniform distribution of lift and drag with respect to its span; as we are assuming in this activity. That said we will focus and discuss only the sectional coefficients of lift and drag as we discuss the aerodynamic performance of airfoils in this laboratory assignment.

Test Configuration

For this laboratory assignment we will be using a rectangular wing with a Clark Y-14 airfoil that spans nearly the full width of the wind tunnel (~12 in). A picture of the model is presented in Figure 3. Note, that we will mount the model in the wind tunnel in a vertical orientation, however this should not influence the wing aerodynamics. Furthermore, we will neglect any three-dimensional effects at the tip and the root of the wing model. Thus we will assume that the flow is two-dimensional or spanwise uniform (i.e. the velocity and pressure distribution will be the same at each spanwise location). As a result we will collect static pressure measurements at only one spanwise location namely at 50% span, in the center of the wind tunnel.

Figure 3: Photograph of the ITLL Clark Y-14 airfoil model with 19 surface pressure ports.

To measure the static pressure distribution on the wing surface the Clark Y-14 wing model is instrumented with 19 static pressure ports uniformly distributed around the surface. These static pressure taps can be visually observed on the model in Figure 3 and their position is defined in the graph found in Figure 4 and by Table 1. We will not measure the absolute static pressure at each of these locations instead we will measure the differential pressure referenced between each port and the freestream static pressure of the wind tunnel (i.e. the difference in pressure between the port and the tunnel freestream). This measurement will simplify our calculation of the pressure coefficient discussed previously.

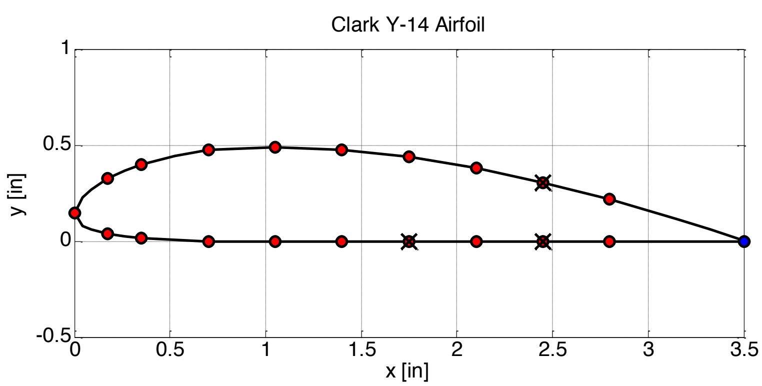

Figure 4: Schematic outlining the position of the surface pressure ports on the Clark Y-14 airfoil model.

As mentioned above, the wing has 19 pressure ports, but our pressure scanner unfortunately only can measure 16 channels at one time. Thus we cannot measure all of the airfoil pressure ports at one time. As a result for this lab we will only consider a subset of the ports available on the Clark Y-14 wing surface and will not connect ports 9, 13, and 15 as documented in Table 1. Finally, to complete the pressure distribution around the airfoil we need a pressure value at the trailing edge (port # 11) of the wing surface, however it would be nearly impossible to instrument a pressure port on the sharp trailing edge of the airfoil. Thus we will need to come up with a theoretical estimate of the differential pressure at the trailing edge. To do this we can extrapolate the pressure at the trailing edge (x = 3.5in) based upon pressure measured at the last two ports on both the upper and lower wing surfaces (upper: p10, p8 and lower: p12, p 14. These linear extrapolations will give us two different estimates for the pressure at the trailing edge, but we know that the pressures from the upper and lower surfaces must converge at the trailing edge. Thus we should average these two pressure estimates to provide one estimate at the trailing edge. We can then incorporate this value into the pressure distribution to complete (and close) the distribution around the Clark Y-14 airfoil. Note: this pressure will change with angle of attack and velocity, thus this procedure will have to be repeated for each of the test cases.

Table 1: Measurement port configuration.

Clark Y-14 Airfoil			Scanivalve Port #
Port #	x [in]	y [in]	
1	0	0.14665	1
2	0.175	0.33075	2
3	0.35	0.4018	3
4	0.7	0.476	4
5	1.05	0.49	5
6	1.4	0.4774	6
7	1.75	0.4403	7
8	2.1	0.38325	8
9	2.45	0.308	**Not Connected**
10	2.8	0.21875	9
11	3.5	0	**Theoretical (Not Physical)**
12	2.8	0	10
13	2.45	0	**Not Connected**
14	2.1	0	11
15	1.75	0	**Not Connected**
16	1.4	0	12
17	1.05	0	13
18	0.7	0.0014	14
19	0.35	0.0175	15
20	0.175	0.03885	16
Wind Tunnel Static Pressure Ring			**Reference**

Measurement Devices

To accurately calculate the lift and drag force coefficients for the Clark Y-14 airfoil several measurements are required. Each will be briefly discussed below. Note that all of the measurements are acquired with the WT2015.vi wind tunnel control program, written in the Labview software language, at a rate of 500Hz for a period of 0.04sec providing a set of 20 measurements at every collection point. These data points are then saved in a collimated data file in a comma separated (.csv) ASCI format. Please refer to the sample data posted on the course website if you have any questions. Note that this data file will store values for all the possible measurements that the wind tunnel data acquisition system can collect, however you may not be using some of these sensors and should thus ignore additional data.

Airspeed Measurements (Pitot-Static Probe Configuration)

Clearly, an accurate measure of the wind tunnel airspeed is required. This is made with the pitot-static probe mounted at the test section entrance and the airspeed differential pressure transducer. Furthermore, the air density is computed using the ideal gas law and measurements of the ambient (atmospheric) pressure and temperature. The details about these measurement devices and methods were fully documented and discussed in the first wind tunnel laboratory assignment and are not repeated here for brevity. Please refer back to this document to address any questions about the airspeed measurement system.

Pressure Distribution Measurements (Scanivalve DSA3217 Multi-Port Pressure Scanner)

Of primary interest in this laboratory assignment is the measurement of the surface pressure distribution on the Clark Y-14 airfoil. These measurements are made on the surface of the wing model at each of the port locations discussed previously. From the wing model each of these ports is plumbed, using flexible tubing, to an associated port on a Scanivalve DSA3217 Multi-Port Pressure Scanner, refer to Table 1 for details about the port configuration. Specifically, the Scanivalve Pressure Scanner incorporates 16 individual pressure transducers that can be sampled nearly simultaneous. Each of these 16 channels has a full scale range of 10in of water and a manufacturer quoted long term accuracy of 0.20% of full scale.

Note: that this device does not measure absolute pressure, instead it measures the differential pressure between each of the designated channels and a reference channel. ***This reference channel should be connected to the wind tunnel static pressure ring to provide each of the surface pressure measurements relative to the wind tunnel static pressure.*** When connected properly this will simplify the calculation of the pressure coefficient discussed previously. If this is not connected then the Scanivalve will supply each of the measurements relative to the room ambient (atmospheric) pressure.

ANALYSIS OF PRESSURE DISTRIBUTION

Several additional steps are required to post-process and analyze the pressure distributions once the differential pressure measurements have been collected. First, as it was discussed above, you need to estimate the pressure at the theoretical trailing edge point (port # 11 in Table 1). This should be done through extrapolating the pressure at the trailing edge from the upper and lower surfaces data. Then you can average these to extrapolations to provide a single estimate of the differential pressure at the trailing edge.

Second, you should compute and plot the pressure distributions in the form of the normalized pressure coefficient versus the normalized chord-wise position, x/c. Figure 5 presents an example of this distribution. This represents the classical form through which pressure distributions around airfoils are compared and allows researchers to easily scale and compare results for similar shaped airfoils at different conditions. Additionally, this will allow you to understand how the flow conditions alter the pressure distribution. For example, what changes do you observed in the pressure distribution when only different wind tunnel speeds are compared? How do changes in the angle of attack alter the distribution? Do you observe any features in the pressure distribution that would indicate the flow has separated at a point on the surface?

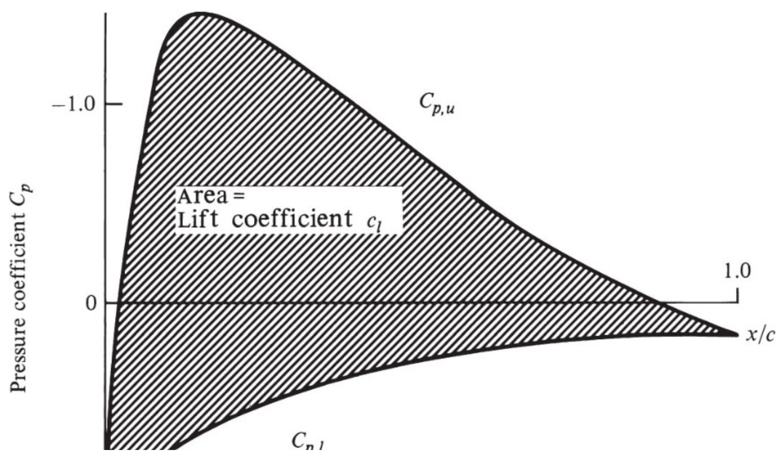

Figure 5: Conceptual graph of the pressure distribution around an airfoil.

Finally, knowing the distribution of pressure on the upper and lower surfaces, we can integrate (sum) all the contributions to determine the net forces perpendicular and parallel to the airflow, i.e., the lift and pressure drag. By repeating this exercise for each of the angles of angles of attack, we can then construct plots to see how the lift and pressure drag vary with the angle of attack, α. Furthermore, we could compare these plots at different airspeeds to understand the influence that the airspeed has on the separation and stalling characteristics of an airfoil. The following section lays out the computational procedure for integrating the pressure distributions.

QUESTIONS TO ADDRESS: *When properly normalized, how does the pressure distribution vary with the airspeed? How do changes in angle of attack alter the pressure distribution? Do you observe any features in the pressure distribution that would indicate the flow has separated at a point on the surface? Finally, were along the airfoil chord is the largest difference in the surface pressure between the upper and lower surfaces? What does this tell you about the pitching moment you can expect on a Clark Y-14 airfoil?*

COMPUTATION OF LIFT AND DRAG

As discussed previously, we must integrate the pressure distribution in order to calculate the lift and pressure drag. Since we sample the pressure at a discrete number of points, we must resort to a numerical integration of the pressure distribution. This procedure described in Figure 6 that shows the airfoil section along with the coordinate systems to be used. Lift and drag (L and D) are defined relative to the oncoming flow direction. The normal and axial forces (n and a) are referenced with respect to the airfoil. It is easiest to perform the force integrations in the airfoil coordinate system (thereby determining n and a) and then transfer these forces to the free-stream coordinate system (giving the desired L and D).

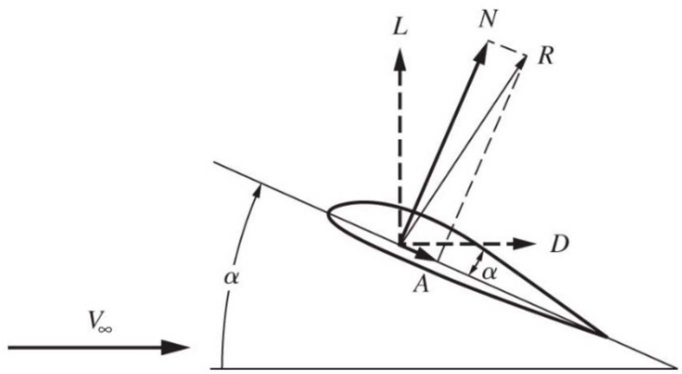

Figure 6: Airfoil orientation and reference systems.

To perform this numerical integration, let the pressure tap locations be denoted as $(x_i,\ y_i)$ and the corresponding pressure be p_i where $i\ =\ 1, 2, \dots, n$. In order to perform the force integrations we will break the airfoil surface into linear elements, each of which span two consecutive pressure taps. Such an element, spanning taps i and $i\ +\ 1$, is shown in Figure 7. We shall assume that the pressure is constant over each element and equal to the average of the pressures at the two end points (where the taps are located).

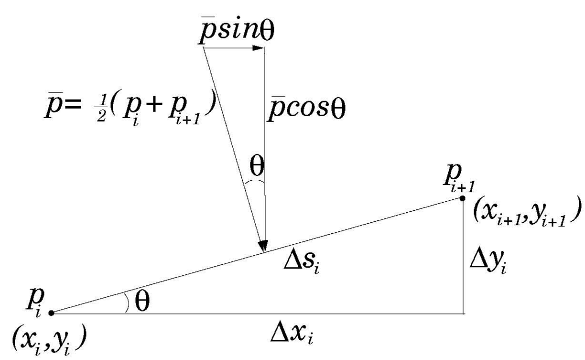

Figure 7: Schematic displaying the elemental components related to the integration of surface pressure between two pressure ports.

According to the sketch, the normal force over the *i^th^* segment is

$$n_i = -\frac{1}{2}(p_i + p_{i+1})cos\theta\Delta s_i.$$

The sketch shows that $cos\theta\ =\ \Delta x_i/\ \Delta s_i$, so that Eq. (3.4) can be simplified to give

$$n_i = -\frac{1}{2}(p_i + p_{i+1})\Delta x_i.$$

A similar development shows that the axial force is given by

$$a_i = \frac{1}{2}(p_i + p_{i+1})sin\theta\Delta s_i,$$

and similar to the normal force

$$a_i = \frac{1}{2}(p_i + p_{i+1})\Delta y_i.$$

Summing over all elements gives the total forces

$$n = -\sum_{i=1}^{n} \frac{1}{2}(p_i + p_{i+1})\Delta x_i$$

and

$$a = \sum_{i=1}^{n} \frac{1}{2}(p_i + p_{i+1})\Delta y_i$$

Note that $\Delta x_i = x_{i+1} - x_i$ and $\Delta y_i = y_{i+1} - y_i$.

As we discussed above, it is convenient to rewrite the force expressions in a non-dimensional form. We accomplish this by adding and subtracting the free-stream pressure (p_∞) from the right hand side and then dividing both sides by the quantity $\frac{1}{2}\rho_\infty V_\infty^2 c$, where *c* is the airfoil chord. The result for the normal force is

$$C_n \equiv \frac{n}{\frac{1}{2}\rho_\infty V_\infty^2 c} = -\sum_{i=1}^{n} \frac{1}{2}\left(\frac{p_i - p_\infty}{\frac{1}{2}\rho_\infty V_\infty^2} + \frac{p_{i+1} - p_\infty}{\frac{1}{2}\rho_\infty V_\infty^2}\right)\frac{\Delta x_i}{c} - \frac{p_\infty}{\frac{1}{2}\rho_\infty V_\infty^2}\sum_{i=1}^{n} \Delta x_i$$

The definition of the pressure coefficient is evident above and the last term vanishes since the summation is taken over a closed contour (the airfoil surface). Making these simplifications, and performing the analogous operations on the expression for the axial force gives

$$C_n = -\sum_{i=1}^{n} \frac{1}{2}\left(C_{p_i} + C_{p_{i+1}}\right)\frac{\Delta x_i}{c}$$

and

$$C_a = \sum_{i=1}^{n} \frac{1}{2}\left(C_{p_i} + C_{p_{i+1}}\right)\frac{\Delta y_i}{c}$$

Notice that these expressions are nothing more than trapezoidal rule approximations to the following two contour integrals, which give the exact normal and axial force coefficients

$$C_n = -\oint C_p \frac{dx}{c}$$

$$C_a = \oint C_p \frac{dy}{c}$$

In order to get the lift and drag, it is necessary to transfer the normal and axial force coefficients to the freestream coordinate system. With reference to Figure 6 it is easy to show that

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

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

If the angle of attack is small we see that the lift is dominated by the normal force and the drag is dominated by the axial force. This procedure and these equations are similar to that presented by Anderson in section 5.7 of our textbook. Furthermore, with a little more analysis we can show that the sectional lift coefficient is fundamentally equal to the difference between the integrated pressure coefficient over the lower surface and the upper surface of an airfoil.

Now that we have defined the procedure for computing the coefficients of lift and pressure drag, we should repeat this computation for each angle of attack and freestream airspeed. We can then construct plots that compare the trends in the lift and pressure drag versus with the angle of attack, α. Furthermore, we could directly compare these same plots at different airspeeds to understand the influence that the airspeed has on the separation and stalling characteristics of an airfoil. Each of these tasks should be accomplished and included in your final report.

QUESTIONS TO ADDRESS: *How do each the lift and pressure drag vary with angle of attack? What is the maximum lift coefficient that this airfoil can generate, and at what angle of attack does that occur? What is the lift produced at zero angle of attack; is it zero? If not why and where is the lift equal to zero? What is the minimum pressure drag and where does that occur? How do these plots and values change at different airspeeds? Do your results match those produced by NACA in 1938 (i.e.* Figure 2*)? If not what are the sources for error in your measurements.*

LABORATORY PROCEDURE

!!!! Take good notes of experimental setup, record data, and consider possible errors in the measurements. You are encouraged to take pictures of the tunnel and your experimental setup with your own devices to help you document your work!!!!

Prelab Tasks:

1. Read the lab assignment in detail and ask questions prior to your assigned measurement period.
2. Review the group assignments spreadsheet to determine the angles of attack and velocities your team has been assigned.
3. From the group assignments spread sheet determine the filename you are required to use for your data. This is important since data will be shared with the entire class.
4. Meet with your group and assign tasks and roles for each student to perform during the wind tunnel measurement period. Develop a plan to ensure you perform your tasks most efficiently.

Procedure:

1. Visually inspect the tunnel and model mounting configuration, compare to your theoretical expectations based upon the details of this assignment and clarify any points of confusion before you get started.
2. Locate the pressure tubes for the pitot-static tube and ensure they are correctly connected to the Airspeed Pressure Transducer such that the wind tunnel control computer can monitor the wind tunnel speed.
3. Locate all of the pressure tubes coming from the airfoil model and ensure that they are correctly connected to the Scanivalve Pressure Scanner such that you are confident in the measurement configuration of the wing model.
4. Find the angle markings on the airfoil base plate, and align the airfoil angle of attack to the first position you were assigned. ***Note: Make sure you have thoroughly tightened the retaining screws to ensure the airfoil is secure and will not move under wind load when the wind tunnel is operating.***
5. Close and lock the wind tunnel door.
6. Open then Start the LabVIEW control program by clicking the white arrow in the top left corner of the window named **WT2015.vi** which is located in the ITLL Courses Folder (**Courses/Fall 2018/ASEN 2002/Airfoil Pressure Lab**).
7. When prompted for the air density type 9999 [kg/m^3].
8. Set your data file path according to the naming convention assigned in the group assignment spreadsheet. All data should be saved in the **"Group Data Save Here/"** directory. ***Note: Be careful not to overwrite another group's data file!!!***
9. Once the LabVIEW VI that controls the wind tunnel and logs the measurements is open and running, zero the airspeed and pressure transducers using the green button on the VI front panel in the bottom left.
10. Type the first velocity that was assigned to your team into the LabVIEW VI and bring the wind tunnel up to speed.
11. Once the wind tunnel has reached equilibrium (typically between 30 and 60 sec), acquire measurements with the pressure transducer by clicking the ***Send Samples to File Button*** in the upper left corner of the VI.
12. Type in the current angle of attack when prompted by the program.
13. Once the data has been collected increase the tunnel speed to your next airspeed setting.
14. Repeat your measurements for each of the velocities at one angle of attack.
15. To change the angle of attack, bring the wind tunnel airspeed back to zero.
16. Once it has settled out open the window, rotate and align the wing model to your next angle of attack, secure the model and close the wind tunnel door.
17. Repeat the pressure measurements again for each velocity assigned to your team and at each angle of attack. ***Note: Be sure to progress through the angles of attack in order, as this information is not saved in the data file. Also be sure to return the wind tunnel to zero airspeed when you change the angle of attack!***
18. Once your data survey is complete, bring the tunnel back to zero, allow time for the tunnel to completely stop, then Stop the VI to save and close your current data file!
19. Take a copy of your data file with you (i.e. usb stick or email) and leave a copy in the **"Group Data Save Here"** folder for the TA to post for the class online.

ANALYSIS OF LABORATORY DATA

You can and should begin the analysis tasks for this lab even before you have collected your own data sets at the wind tunnel. Several of these items can be accomplished before your assigned testing period as they do not require experimental data. For example sample data collected by the professor, TAs and Laboratory Staff has been posted on the website. You can begin working on your matlab scripts that will load and process the data files. The computational procedure for computing the lift from the pressure distribution is very involved and should be started early! You should work on these tasks together with your group during your lab session. Use the background material, *Questions to Address* in this document, and the report grading rubric as a guide for your analysis.

It is recommended that you use Matlab to analyze and plot all of your experimental data. You can use the sample data files posted on the course website and your classmate's submitted data files to begin the process of building your data loading and post-processing scripts, even before you acquire your own data. Note that the data files collected with the wind tunnel are all stored in the same format as a csv file where each column contains a different measurement and the rows are the individual data points collected, refer to the data measurement section above for more detail. To develop a complete understanding of how the lift and pressure drag vary with angle of attack and airspeed you will need to also analyze your classmates' data! Your data is not complete on its own; you will need your classmates' data to complement your own in your analysis. Specifically do to the time constraints you are only collecting pressure measurements at three angles of attack. When you combine your results with your classmates' data you can reconstruct the lift and pressure drag distributions versus angle of attack from -15º to 15º for 1º increments. You should begin analyzing and processing these results early as the data files are available.