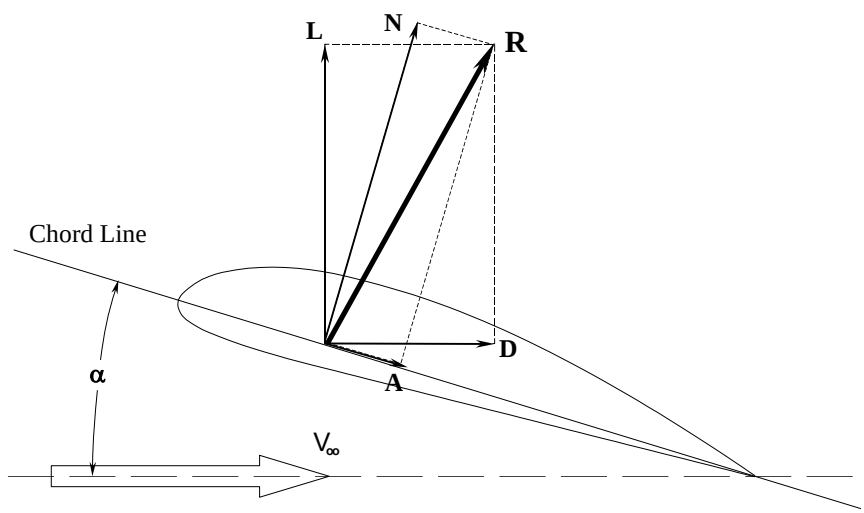
Laboratory

7

Airfoil Lift Determination
Observations of the Distribution of Pressure

Introduction

The imbalance of pressure forces over the upper and lower surfaces of a wing yields a net vertical force. This is mainly because the generation of lift creates lower pressures above the airfoil and higher pressures below. In this laboratory, we will explore how the distribution of pressure affects the airfoil lift. The results will reflect upon the importance of the leading edge region in the development of efficient lift.



• Figure 5. Forces acting on an airfoil

Airfoil Lift and the Lift Coefficient

Consider the airfoil section in Figure 5, where the angle of attack with respect to the incoming flow is labeled as α (also called AOA or the incidence angle). The freestream velocity is labeled as V_∞. The distribution of pressure around the airfoil yields a net resultant aerodynamic force; this force is labeled as R and it generally increases with α. The resultant force can be decomposed using two different orthogonal axis systems. The first of these is relative to the airfoil chord line, the imaginary line extending through the leading edge and the trailing edge. The force that is parallel to the chord line is known as the *axial force* and the perpendicular force is known as the *normal force*. These are labeled as A and N, respectively, in Figure 5.

The second orthogonal axis system is oriented with the incoming wind direction, also known as the relative wind vector. Using this system, a more convenient force decomposition involves the drag (D), which is <u>parallel</u> to the relative wind, and the lift (L), which is <u>perpendicular</u> to the relative wind. From trigonometry, we obtain the relationship between the two axis systems; we are presently concerned with only the lift. The lift equation becomes:

$$L = N\cos(\alpha) - A\sin(\alpha) \tag{7.1}$$

However, this relation can be simplified somewhat. In practice, the magnitude of N is much larger than the magnitude of A, with a ratio of anywhere from 60:1 to 130:1 over the range of this experiment. Combined with the fact that small airfoils can endure only relatively small incidence angles, perhaps only as high as 12° or so, we realize that the term $\cos(\alpha)$ is much larger than the term $\sin(\alpha)$. This leads to a fairly accurate approximation for the lift that can be easily obtained in a simple laboratory experiment using:

$$L \approx N\cos(\alpha) \tag{7.2}$$

Then, in order to calculate the lift coefficient, we need to divide the lift force by the freestream dynamic pressure, q_∞, and the wing planform area, S. If consistent units are chosen, the following equation will yield a *dimensionless* lift coefficient:

$$C_L = \frac{L}{q_\infty S} \approx \frac{N\cos(\alpha)}{q_\infty S} = C_N \cos(\alpha) \tag{7.3}$$

It can be seen that the lift coefficient depends on the so-called normal force coefficient, C_N. However, for an infinite wing, lower-case subscripts are used to designate two-dimensional section characteristics. Thus for this experiment:

$$C_\ell \approx C_n \cos(\alpha) \tag{7.4}$$

Normal Force Coefficient

Thus, to calculate the lift coefficient, we require the lift force. And to calculate the lift force, we require the normal force. Now all that is needed is an accurate measurement of the normal force, N. This can be an involved derivation, so only the principal results will be presented here. You should consult section 5.7 of your text (pp. 252-256) for more detail.

In essence, the increment to the perpendicular normal force is the difference of the pressure forces acting on differential areas on the lower and upper surfaces, respectively. The pressure force on lower surface acts in the positive direction and the pressure force on the upper surface acts in the negative direction. This is written as:

$$dN = dF_L - dF_U = p_L(b \cdot dx) - p_U(b \cdot dx) \tag{7.5}$$

Note that p_L and p_U are the lower and upper surface pressures, respectively. The area these pressures are acting on is $b \cdot dx$, where b is the span of the airfoil section and dx is the chordwise increment. Simplifying, the equation then becomes:

$$dN = b(p_L - p_U)dx \tag{7.6}$$

Dividing by the dynamic pressure and the wing area results in the contribution to the normal force coefficient. In this instance, the rectangular wing area simply becomes the product of the chord and span ($S = cb$), such that:

$$dC_n = \frac{dN}{q_\infty S} = \frac{b(p_L - p_U)dx}{q_\infty cb} = \frac{(p_L - p_U)}{q_\infty} \cdot d\left(\frac{x}{c}\right) \tag{7.7}$$

But this can be simplified further when considering the definition of pressure coefficient:

$$C_p \equiv \frac{p - p_\infty}{q_\infty} \tag{7.8}$$

Equation (7.7) is simplified by subtracting and adding the freestream pressure, p_∞, and substituting equation (7.8):

$$dC_n = \frac{(p_L - p_\infty) - (p_U - p_\infty)}{q_\infty} \cdot d\left(\frac{x}{c}\right) = \left(C_{p_L} - C_{p_U}\right) \cdot d\left(\frac{x}{c}\right) \tag{7.9}$$

To get the total normal force coefficient over the entire airfoil (from the leading edge to the trailing edge), we must integrate over the chord length. This means the value of x/c goes from 0 to 1:

$$C_n = \int_0^1 \left(C_{p_L} - C_{p_U}\right) d\left(\frac{x}{c}\right) = \int_0^1 C_{p_L}\, d\left(\frac{x}{c}\right) - \int_0^1 C_{p_U}\, d\left(\frac{x}{c}\right) \tag{7.10}$$

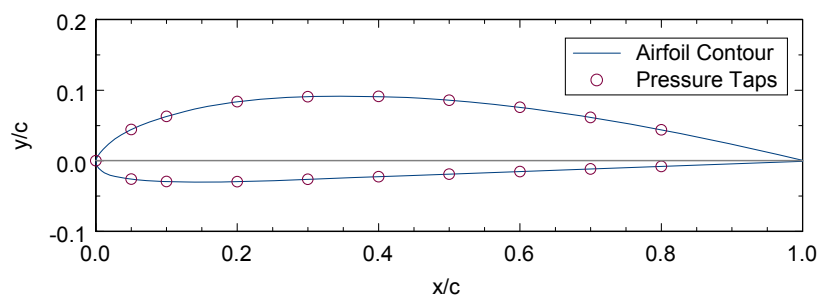
Therefore, we can get the normal force coefficient by *integrating* the plot of $C_{p_L} - C_{p_U}$ as a function of x/c. Or, more easily, we can get the area between the C_{p_L} and C_{p_U} curves. Unit cancellation shows us that if we use the dimensionless quantities for C_p and x/c, then the coefficient C_n will also be dimensionless.

The Clark-Y14 Airfoil

The airfoil used in this laboratory is one of the earliest airfoil designs, the Clark-Y14. The Clark-Y14 is a cambered airfoil with a maximum deviation from the chord line of 4.6% of the chord length. The thickness-to-chord ratio is about 0.14. These parameters are summarized in Table 7. Figure 6 shows the airfoil contour on a dimensional plot. Note how the mean line between the upper and lower surfaces is curved, with a maximum deviation of about 0.16" from the chord line.

• Table 7. Geometry details of the Clark-Y14 airfoil

Chord (inches)	3.50
Span (inches)	10.75
Thickness	14.0%
Camber	4.6%



• Figure 6. Clark-Y14 airfoil contour and pressure tap locations

The Inclined Manometer Bank

Also depicted in Figure 6 are the locations of the surface pressure taps, from which the airfoil pressure distribution will be measured. There are 18 pressure taps that are connected to a manometer bank with flexible tubing; the dimensionless coordinates of these pressure taps are listed in Table 8. The manometer bank is shown in Figure 7. Through the flexible plastic tubing, the surface pressures will be transmitted to the bank for ease of measurement. **YOU MUST PAY ATTENTION TO UNITS IN THIS EXPERIMENT**, because the manometer bank is *inclined* at an angle of 30° and thus measures a "skewed" pressure difference. In order to get a true vertical column of fluid (and thus an accurate pressure measurement), we have to apply the equation:

$$\Delta h = \Delta l \cdot \sin(30) = 0.50 \cdot \Delta l \tag{7.11}$$

• Table 8. Surface pressure tap locations

x/c	Upper Surface		Lower Surface	
	No.	y/c	No.	y/c
0.00	1	0.0000	1	0.0000
0.05	2	0.0443	3	-0.0261
0.10	4	0.0630	5	-0.0294
0.20	6	0.0839	7	-0.0297
0.30	8	0.0907	9	-0.0263
0.40	10	0.0912	11	-0.0226
0.50	12	0.0859	13	-0.0190
0.60	14	0.0758	15	-0.0153
0.70	16	0.0614	17	-0.0116
0.80	18	0.0439	19	-0.0079

In equation (7.11), Δl is the length of the *inclined* manometer column and Δh is the vertical height difference. Furthermore, the fluid in the manometer tubes is not water, but alcohol. The alcohol used has a specific gravity of 0.811, so this must be factored into the equation:

$$\Delta h_{water} = \Delta h_{alcohol} \cdot 0.811 = \left(0.50 \cdot \Delta l_{alcohol}\right) \cdot 0.811 \tag{7.12}$$

Thus we can convert the pressure measurements from units of inches alcohol to units of inches H_2O. Now Δh, which is really a pressure measurement, can be converted to the units required for the data reduction (NOTE: 1 atmosphere=406.78 inches H_2O).

However, the manometer bank is actually showing you the <u>gauge</u> pressure, not the <u>absolute</u> pressure on the airfoil surface. It is called gauge pressure because it is referenced to the atmospheric pressure, and is written as **psig** (lbs/in², gauge). This is in contrast to **psia** (lbs/in², absolute). Since the reservoir of the manometer bank is open to atmospheric pressure, the pressure indicated will be the gauge pressure. So instead of indicating p_L and p_U, it is really indicating $p_L - p_{atm}$ and $p_U - p_{atm}$. You must take this into consideration when calculating surface pressures and pressure coefficient values. To get the pressures in absolute terms (psia), convert the Δh value from in. H_2O to lbs/in² and then add the atmospheric pressure (in lbs/in²) for each data point:

$$p = \Delta h_{water} \cdot \left(\frac{14.7 \quad psi}{407 \quad "H_2O}\right) + p_{atm} \tag{7.13}$$

Helpful Note: The tops of the manometer tubes are connected to the surface pressure taps and the bottoms are plumbed into the fluid reservoir. Therefore, a <u>positive</u> deflection in the fluid column actually represents a <u>negative</u> gauge pressure, $p - p_{atm}$.

Procedure

Follow these steps to operate the 12x12" Aerolab Educational Wind Tunnel #1 for this laboratory:

1) Apply AC power to the control computer of the Educational Wind Tunnel (EWT). Let the electronics "warm up" for at least 30 minutes before operating the wind tunnel.
2) Record the barometric pressure and the room temperature.
3) Check the manometer bank to ensure that all tubes are reading zero with the air speed at zero.

4) Hit the button to “zero out” the balance readouts for velocity, normal force, axial force, and pitching moment. You will only be using the computer to set the velocity in the test section.

5) After clearing the area immediately in front of the inlet and behind the exhaust, turn on the VFD (variable frequency drive) of the motor controller. By using the slider control, bring up the tunnel speed manually to 50% speed. Under the “Setpoint” window, type in 80 mph and enable the automatic speed control by clicking the “Auto Control” button.

6) Adjust the airfoil section to zero degrees AOA. When the airspeed stabilizes, record the height of the alcohol columns on the manometer bank for tubes 1-19. Record the height of the reference tube (labeled as “ R ” on the metal strip), which corresponds to the test section static pressure, p_∞. You should notice that most of the raw pressure data will be <u>negative</u>. Note that these pressures should be recorded in the units of “inches alcohol.” Use the table on page 16 as your datasheet.

7) Repeat steps 6-7 for AOA=5°, 10°, and 15°.

8) When complete, shut down the tunnel by disabling the auto control and reducing power to 0%. Turn off the VFD of the motor controller.

• Figure 7. Alcohol manometer bank

Data Reduction

The data reduction should involve the following calculations for all the wind tunnel tests run in your class section:

- Calculate the air density.
- Convert the pressures from inches alcohol (gauge) to lbs/in² (absolute).
- To calculate the test section dynamic pressure, don't use the displayed test section velocity because it's not the true value. Instead, take the reference pressure and multiply by the following correction factor:

$$q_\infty = 0.965\left(p_{atm} - p_\infty\right)$$

- Calculate the pressure coefficients C_{p_L} and C_{p_U} from the leading edge to the trailing edge.
- For each angle tested, plot C_{p_L} and C_{p_U} against x/c on the same graph. To close up your graph, set the pressure at the trailing edge equal to the reference pressure (i.e. where x/c = 1.00, C_p = 0.00). Remember that this reference pressure is actually the test section static pressure.
- At each AOA tested, calculate the value of the normal force coefficient by determining the area between the curves. You may use the data values to calculate individual trapezoidal areas, or you may plot the data on a grid and count the number of blocks bounded by the two curves. Whichever method you chose, you must show all your work!
- At each AOA tested, calculate the value of C_n and C_ℓ. Tabulate these values.
- Compare your values with the following 2D theoretical results[11] for the Clark-Y14. These results were generated at a Reynolds number of 220,000 and a Mach number of 0.1:

- Table 9. Theoretical results for a Clark-Y14 airfoil section

α (deg)	C_ℓ
-5	-0.127
0	0.541
5	0.997
10	1.413
15	1.497

Deliverables

Although the data were gathered in teams, each write-up must be done and submitted on an individual basis. You must hand in the following deliverables to complete this laboratory:

- A formal **CONCLUSIONS** write-up for this experiment. This should be between one and three brief paragraphs in length. Remember that is should be a brief summary of what might go into a full-blown results section. Some items to consider:
- Is the method an effective experimental exercise? Why or why not?

[11] These results were obtained using the XFOIL code compiled for Microsoft Windows.

- What was the minimum percent error? What are some factors that might have affected your results?
- Can you isolate a region along the airfoil chord that is primarily responsible for the generation of lift? What is the range of x/c values?
- What was the Reynolds Number for your airfoil test? Use the chord length as the representative length in the calculation. How might this influence the flow over the Clark-Y14 airfoil?
- An appendix with your raw data table
- Plots of C_p (both upper and lower) as functions of x/c for each AOA
- A table of C_n and C_ℓ values for each AOA, and percent error from the theoretical 2D results
- Sample calculations for your data that are used to obtain your results

• Table 10. Raw data table for pressure distribution experiment

Date & Time: _____ Barometric Pressure: _____

Test Section Airspeed: _____ Room Temperature: _____

Port No.	x/c	Manometer Column Readings,[12] $\Delta\ell$ (inches alcohol)			
		α_1 =	α_2 =	α_3 =	α_4 =
R					
1	0.00				
2	0.05				
3	0.05				
4	0.10				
5	0.10				
6	0.20				
7	0.20				
8	0.30				
9	0.30				
10	0.40				
11	0.40				
12	0.50				
13	0.50				
14	0.60				
15	0.60				
16	0.70				
17	0.70				
18	0.80				
19	0.80				
"20"[13]	1.00				

[12] **NOTE**: All manometer readings should be negative.
[13] There is no pressure port at the trailing edge; this pressure is assumed to be the same as "**R**."