AERODYNAMICS LAB 2 – AIRFOIL PRESSURE MEASUREMENTS NACA 0012

1 Objective

To use pressure distribution to determine the aerodynamic lift and drag forces experienced by a NACA 0012 airfoil placed in a uniform free-stream velocity.

2 Materials and Equipment

- UAH 1-ft x 1-ft open circuit wind tunnel
- Traversing mechanism
- Pitot-static probe
- Molded epoxy NACA 0012 airfoil section with a 4-inch chord and an array of 9 pressure taps along its upper surface
- Digital pressure transducer
- Data Acquisition (DAQ) Box

3 Background

3.1 Airfoil Lift and Drag

We can determine the net fluid mechanic force acting on an immersed body using pressure measurements on the surface and in the viscous, separated wake. The net force can be resolved into two components: the **lift** component, which is normal to the freestream velocity vector; and the **drag** component, which is parallel to the freestream velocity as shown on Figure 1.

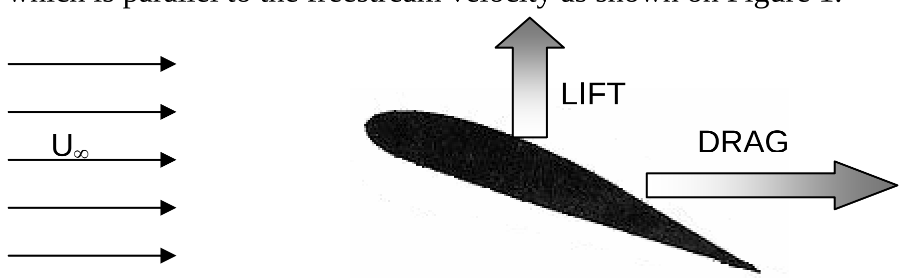

Figure 1 - Aerodynamic Force on an Airfoil

We often express these forces in non-dimensional coefficient form

$$C_F = \frac{F}{\left(\frac{1}{2}\rho V_\infty^2\right) A_{REF}}, \tag{1}$$

where F can be the lift (L) or drag (D) force, and A_{REF} is a specified reference area. For two-dimensional bodies the force is per unit span (or width), or the area is determined with a unit span.

3.2 Governing Equations

Ideal Gas Law
At standard conditions, air behaves very much like an ideal gas (the intermolecular forces are negligible). As a result, we can express relation between the pressure, p, the density, ρ, the temperature, T, and a specific gas constant, R (for air, R = 287 J/(kg K)), as

$$p = \rho RT \ . \tag{2}$$

Sutherland's Viscosity Correlation
At standard conditions, an empirical relationship between temperature and viscosity given by the Sutherland correlation

$$\mu = \frac{bT^{1/2}}{1 + S/T} \ , \tag{3}$$

where $b = 1.458 \times 10^{-6} \ kg/(m \cdot s)$ and $S = 110.4 \text{ K}$.

Bernoulli's Equation
For a steady, incompressible, inviscid, irrotational fluid flow, a relation between p, the ***static pressure*** (due to random molecular motion of the fluid molecules), $\frac{1}{2}\rho V^2$, the ***dynamic pressure*** (due to the directed motion of the fluid), and p_o, the ***total/stagnation pressure*** (pressure you would sense if the fluid flow was isentropically brought to rest), called Bernoulli's equation, can be derived as

$$p_o = p + \frac{1}{2}\rho V^2 = const. \tag{4}$$

Bernoulli's equation can be used to determine the velocity of an incompressible fluid flow.

3.3 Similarity Parameters

The bodies tested in the wind tunnel are generally scale models of a full size prototype. As a result, we must introduce similarity parameters that will allow us to perform a study of dimensional analysis and similitude.

Reynolds Number
The Reynolds number is the ratio of inertia forces to viscous forces. 'Low' Reynolds number flows tend to be dominated by viscosity and thus exhibit laminar boundary layers, while 'high' Reynolds number flows tend to exhibit turbulent boundary layers. The Reynolds number can be expressed as

$$\mathrm{Re} = \frac{\rho V c}{\mu} \ , \tag{5}$$

where ρ and μ are, respectively, the density and the viscosity of the fluid, V is the flow velocity, and c is a characteristic dimension of the body.

3.4 Pressure Coefficients

The pressure coefficient can be expressed

$$C_p = \frac{p - p_{REF}}{\left(\frac{1}{2}\rho V^2\right)_{REF}}. \qquad (6)$$

The pressure coefficient is thus the difference in the local pressure and a reference pressure divided by the reference dynamic pressure. Typically, the freestream values far ahead of the body (denoted by the subscript '∞') are used for the reference conditions.

3.5 Lift Coefficient

Consider the pressure and shear stress distributions along the surface of an immersed body. We can divide the surface into small, elemental areas and resolve the contributions to lift and drag on each area (see Figs. 1.15, 1.16, 1.17, and 1.18 of Ref. 1). The net lift and drag forces are obtained by summing up these elemental contributions (i.e., integrating). Empirical results indicate that we can generally neglect the shear stress contribution to the lift and only consider the contributions of pressure on the upper and lower body surfaces (as shown on Figure 2).

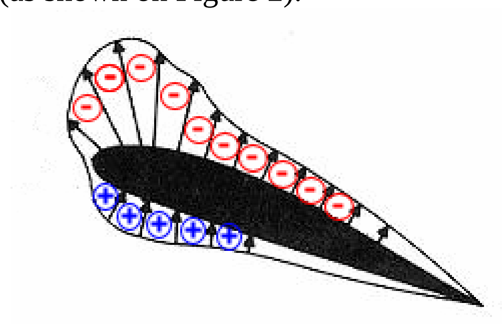

Figure 2 - Pressure Distribution Around an Airfoil

Using this approach for a two-dimensional (or infinite span) body, a relatively simple equation for the lift coefficient can be derived

$$C_l = \cos\alpha \int_{x/c=0}^{x/c=1.0} \left(Cp_{lower} - Cp_{upper}\right) d\left(\frac{x}{c}\right), \qquad (7)$$

where α is the angle of attack, c is the body chord length, and the pressure coefficients (*Cp*s)are functions of the normalized length x/c. Note that we use a lower case "*l*" to designate a two-dimensional body or force per unit span.

3.6 Drag Coefficient

For smooth streamlined bodies (such as an airfoil), the drag is predominantly due to shear stress. The surface integration technique requires knowledge of the shear stress distribution along the surface, which may be difficult to obtain experimentally. In this case, we can estimate the drag of the body by comparing the momentum in the air ahead of the body to the momentum behind the body.

The total momentum loss can be equated to the drag of the body by application of a momentum integral analysis (e.g., Chapter 3 of Ref 2). A Pitot-static probe can be traversed along vertical planes ahead and behind the body to determine the profiles of local dynamic pressure and associated flow momentum. In Ref. 3, an equation is derived for the drag of an immersed body based on this dynamic pressure profile in the separated wake. The resultant equation is given by

$$C_d = \frac{2}{d}\int_{Y_1}^{Y_2}\left[\sqrt{\frac{q}{q_\infty}} - \frac{q}{q_\infty}\right]dY\ , \qquad (8)$$

where q and q_∞ are the local and freestream values of dynamic pressure, d is the cylinder diameter, and Y_1 and Y_2 are the beginning and ending coordinates of the vertical pressure probe traverse. Proper values of q are only obtained if the wake has returned to the tunnel static pressure, p_∞, and not the local static pressure near the body. Performing the pressure traverse several chord lengths behind the body rectifies this problem.

4 Procedure

4.1 Determination of the Lift Coefficient From Surface Pressure Measurements

The coefficient of lift can be obtained by integrating the measured pressure profile around the airfoil using Eq. (7). The test model is a molded epoxy NACA 0012 airfoil section with a 4-inch chord and an array of 9 pressure taps along its upper surface. The airfoil spans the test section width (i.e., infinite span) and can be set at various angles of attack to the flow. The airfoil pressure tap locations are provided in table 1.

Table 1 - Airfoil static pressure tap locations

Port #	Distance Along the Chord From Leading Edge (mm)
1	4
2	10
3	20
4	30
5	40
6	50
7	60
8	70
9	80

The following procedure is used to determine the surface pressure distribution:

1. Turn on the data acquisition box (DAQ) and allow approximately ten minutes for the pressure transducer to warm up. Make sure the multi-colored tube bundle is attached to the airfoil pressure tap stems and the DAQ.

2. Measure the lab static pressure and temperature with the digital barometer and thermometer. Calculate the air density ρ and viscosity μ using Eqs. (2) and (3).
3. Calculate the tunnel dynamic pressure to attain an airfoil chord Reynolds number (Re_c) of approximately 250,000.
4. Set the pressure port selector switch to 0, and then ZERO the pressure transducer.
5. Place the Pitot-static probe at the farthest upstream position and approximately 1.5 inches below the test section roof.
6. Using the scribed index line on the edge of the airfoil wall plug and the degree scale surrounding the wall port, place the airfoil at an angle of attack of $\alpha = 0°$.
7. Turn on the wind tunnel and adjust the speed control until the pressure transducer reads the calculated dynamic pressure setting for $Re_c = 250{,}000$.
8. Manually turn the port selector dial through the nine pressure ports. Pause at each port and record the pressure differential, $\Delta p = p_{port} - p_{\infty}$.
9. Turn off the air flow and see if the pressure transducer returns to zero.
10. Repeat steps (6) - (8) for α = 4°, -4°, 10°, and -10°. The pressures for negative α values represent the lower surface of the airfoil at positive α values.

4.2 Determination of the Lift and Drag Coefficients From Wake Pressure Measurements

By measuring the velocity profiles in the wake and using conservation of linear momentum, the drag coefficient on the cylinder can be determined using Eq. (8). The experimental set up will be as shown in Figure 3 where wake measurements are obtained a short distance behind the airfoil.

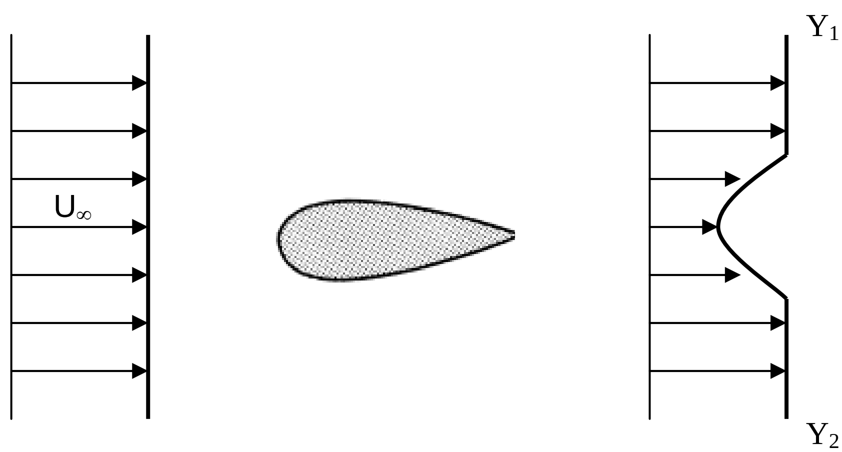

Figure 3 - Wake Pressure Distribution

The following procedure is used to determine the wake pressure distribution:

1. Place the Pitot-static probe at an axial location of approximately one chord length behind the airfoil.
2. Traverse the probe vertically across the airfoil wake, recording the dynamic pressure at discrete Y-locations. Use enough points to adequately resolve the relatively small wake and dynamic pressure minimum.
3. Repeat for angle of attack: α = 4°and 10°.

5 Laboratory Report

1. Calculate the actual airfoil chord Reynolds number.
2. Calculate the *Cp* distribution on the airfoil surface at each angle of attack. There should be a stagnation point (C_p = 1.0) at the trailing edge (x/c = 1). A stagnation point will also exist near the airfoil nose. This point is at x/c = 0 for α = 0°. For positive angles of attack the stagnation point is on the lower surface. For these cases we can assume that the C_p at x/c = 0 is approximately the average of the C_p values at port 1 on the upper and lower surfaces (positive and negative angles of attack). As indicated on the data sheet, include these values in your data set.
3. On one composite graph plot the negative of the pressure coefficient (**-***Cp*) versus x/c for each angle of attack. This will put the upper surface curves on the top. Discuss the effects of α on the airfoil pressure distribution.
4. Use the *Cp* values (including the x/c = 0 and 1 points) from step (2) and Eq. (7) to calculate the airfoil lift coefficient at the three angles of attack.
5. On one composite graph, plot the dynamic pressure values (q) from the airfoil wake traverses versus Y-location. Discuss the effects of α on the wake location and width.
6. Use the q values and Eq. (8) to calculate the airfoil drag coefficient at each angle of attack.
7. Plot the C_l and C_d values calculated in steps (4) and (6) vs. α. Discuss the effects of α on the lift coefficient. Estimate the lift curve slope and comment on your value. Discuss the effects of α on the drag coefficient.

References

[1] Anderson, J. D., ***Fundamentals of Aerodynamics***, 4th ed., McGraw Hill, 2007, pp. 19-16.
[2] Barlow, J. B., Rae, W. H., Jr., and Pope, A., ***Low-Speed Wind Tunnel Testing***, 3rd ed., Wiley-Interscience, 1999, pp. 176-179.

MAE 449 – AERODYNAMICS LAB 2 – DATA SHEET

Airfoil Dimensions

Chord length: c = 0.1016 m
Span width: b = 0.3016 m

Lab Conditions :

P_{lab} = ______________ [Pa]

T_{lab} = ________________ [°C]

ρ_{lab} = ________________ [kg/m^3]

μ_{lab} = ________________ [kg/(m.s)]

Surface Pressures	q_∞ **[Pa] =**					
Port #	***x/c***	$\Delta p = p_{port} - p_\infty$				
		α = 0°	**α = 4°**	**α = -4°**	**α = 10°**	**α = -10°**
0	**0**	**Stagnation Point : Cp=average(Cp_{upper}, Cp_{lower})**				
1	**0.0394**					
2	**0.0984**					
3	**0.1969**					
4	**0.2953**					
5	**0.3937**					
6	**0.4921**					
7	**0.5906**					
8	**0.6890**					
9	**0.7874**					
10	**1**	**Stagnation Point : Cp=1**				

Wake Pressures		q_∞ **[N/m^2] =**			
α **= 0°**		α **= 4°**		α **= 10°**	
Y [mm]	q **[N/m^2]**	**Y [mm]**	q **[N/m^2]**	**Y [mm]**	q **[N/m^2]**