Experiment 4 : Airfoil C_p Distribution and Lift

AS2510 Low Speed Lab

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Aim

To determine the C_p distribution over an airfoil, determine the lift generated by the airfoil and its C_L

Apparatus

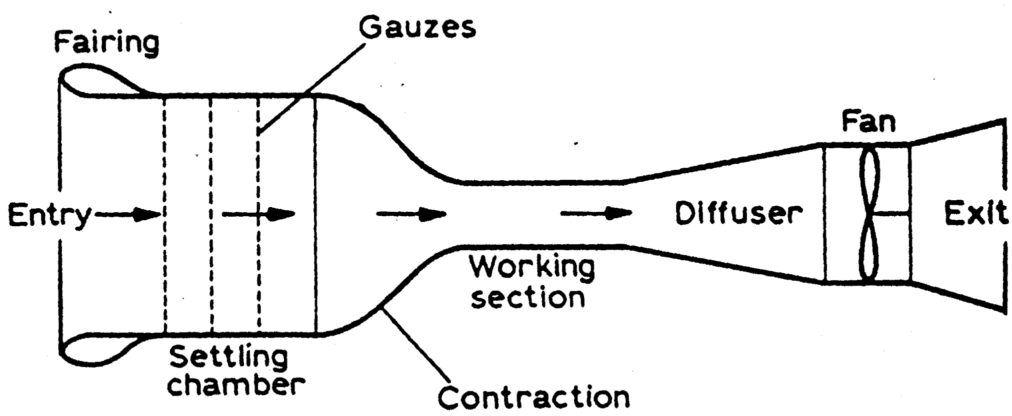


Figure 1: Schematic of open return wind tunnel used in experiment

The apparatus is similar to the one depicted above. The tunnel cross section in the working section is 150 mm x 150 mm through out. A NACA0015 Airfoil model (with chord length c = 65 mm) is kept in the working section with probes attached to its upper surface only. Since NACA0015 is a symmetric airfoil section, the measurements at are taken for two angles of attack with equal magnitude and opposite sign. This enables us to obtain the pressure distribution on both, the upper and the lower, surfaces.

In our experimental set-up, there are 10 ports located at distances d_i from the leading edge :

Port no.	1	2	3	4	5	6	7	8	9	10
d_i(mm)	0	3	5	7	9	22	29	36	43	50
x/c	0	0.046	0.077	0.108	0.138	0.338	0.446	0.554	0.662	0.769

Table 1: Position of Pressure Probes on the Airfoil Section

To get measurements from this apparatus we use a flow meter.

We use a static pressure tap to measure the static pressure at different points in the flow. We also use a pitot tube to measure and verify the stagnation pressure. These are recorded with the help of a flow meter.

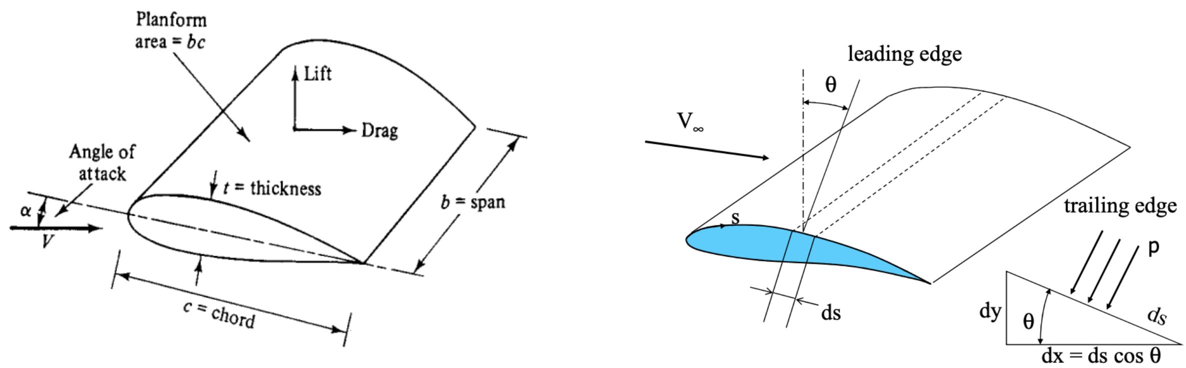
Since the source of the flow is the atmosphere, we know the stagnation pressure to be equal to the atmospheric pressure. The head loss in the low-speed flow is negligible as compared to the initial stagnation pressure and hence we use the atmospheric pressure as the stagnation pressure.

Principle

A body immersed in a flowing fluid is exposed to both pressure and viscous forces. The sum of the forces that acts normal to the free-stream direction is the lift, and the sum that acts parallel to the free-stream direction is the drag.

Because the velocity of the flow over the top of the airfoil is greater than the free-stream velocity, the pressure over the top is negative. This follows directly from the application of Bernoulli's equation. Similarly the velocity along the underside of the airfoil is less than the free-stream velocity and the pressure there is positive.

Hence, both the negative pressure over the top and the positive pressure along the bottom contribute to the lift. In this experiment, the lift force, L, on the airfoil will be determined by integration of the measured pressure distribution over the airfoil's surface.



(a) Geometric and Dynamic parameters of airfoils (b) Obtaining Lift from Pressure Distribution

Figure 2: Airfoil Section Characteristics

The pressure coefficient is calculated using

$$C_{p_i} = \frac{p_i - p_\infty}{0.5\rho_\infty V_\infty^2}$$

We use discrete integration to calculate the Lift. The lift coefficient is then calculating using

$$C_L = \frac{L}{0.5\rho_\infty V_\infty^2 bc}$$

Procedure

1. Set-up the apparatus with a steady flow
2. Measure the pressure at different location using the pressure probes on the apparatus
3. With the pressure measurements, stagnation pressure calculate the Pressure Coefficient.
4. Plot the variation of Pressure Coefficient along the chord of the airfoil.

Results

Calculations & Results

Atmospheric Pressure (ρ_∞) : 101325 Pa
Density of Air : 1.2754 kg/m^3

Angle of Attack = 2 degrees
V_∞ = 10 m/s

Pitot Pressure Measured : 101321 Pa (4 Pa below atmospheric pressure)

% change in Stagnation pressure = (4 Pa / 101325 Pa) $\times$ 100 $\approx$ 0.004%

Thus, we can use the atmospheric pressure as the stagnation pressure.

To obtain the static pressure from the experimental readings, we use the following :

$$static\ pressure = absolute\ pressure = atmospheric\ pressure + gauge\ pressure$$

$$gauge\ pressure = measurement \times conversion\ factor$$

After which, the pressure coefficient is calculated using

$$C_{p_i} = \frac{p_i - p_\infty}{0.5\rho_\infty V_\infty^2}$$

	gauge pressure (mm H_2O)	C_{p_u}	gauge pressure (mm H_2O)	C_{p_l}
Port 1	-0.8	-0.124	-2.6	-0.400
Port 2	-9.6	-1.476	-4.1	-0.630
Port 3	-11.9	-1.830	-4.3	-0.661
Port 4	-12.2	-1.876	-7.6	-1.169
Port 5	-11.2	-1.722	-6.3	-0.969
Port 6	-10.5	-1.615	-8.2	-1.261
Port 7	-10.2	-1.569	-8.1	-1.246
Port 8	-10	-1.538	-8	-1.230
Port 9	-9.9	-1.522	-8	-1.230
Port 10	-9.5	-1.469	-7.4	-1.128

Table 2: Experiment Readings for Upper Surface and Lower Surface

C_p variation along the chord

The plot below shows the variation in free-stream velocity along the apparatus.

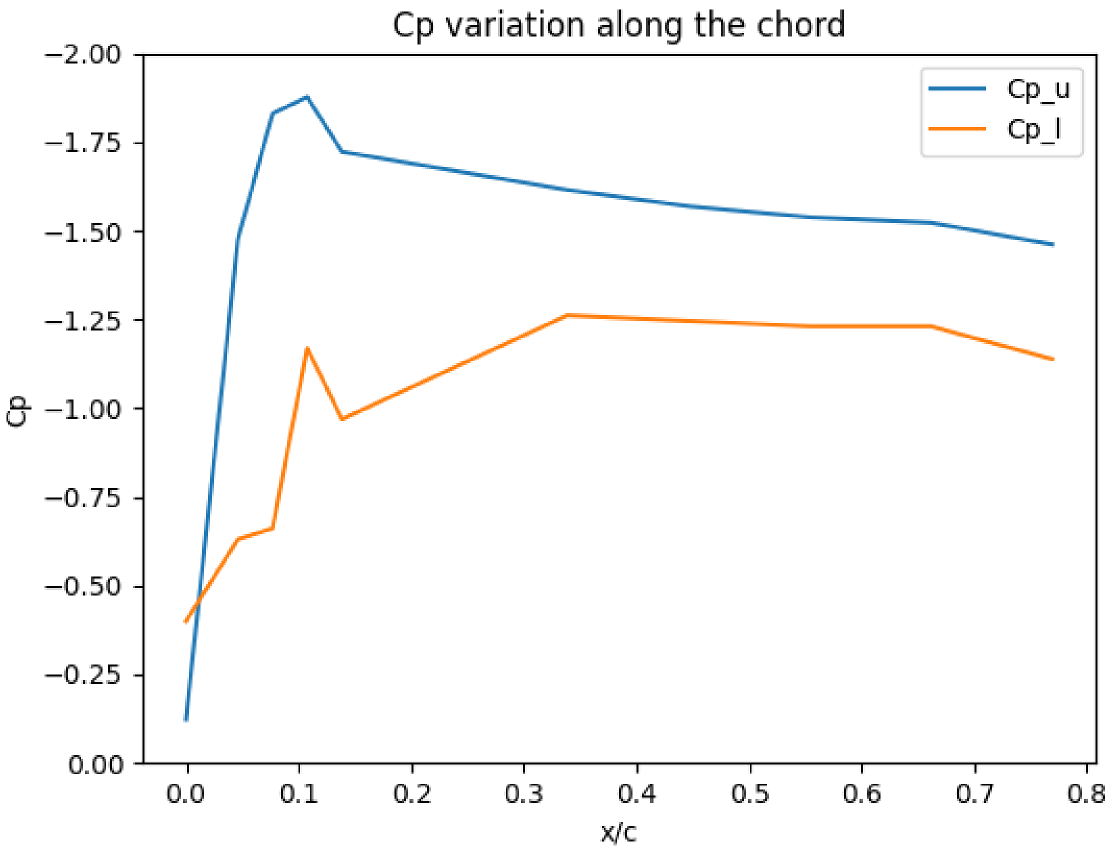


Figure 3: Plot showing the variation of C_p along the chord

Calculation of Lift and C_l

To find the Lift, we need co-ordinates for the NACA0015 airfoil which can be found using the formula for symmetric airfoils

$$\frac{y_t}{c} = \frac{t}{0.2}[0.2969\sqrt{\frac{x}{c}} - 0.1260(\frac{x}{c}) - 0.3516(\frac{x}{c})^2 + 0.2843(\frac{x}{c})^3 - 0.1015(\frac{x}{c})^4]$$

where c is the chord length and t is the relative thickness as a fraction of the the chord.

For NACA0015 airfoil in our case, c = 65mm, t = 0.15. Since we are dealing with x/c, it is convinient to calculate y/c to find the angle of inclication of the panel w.r.t the chord.

For the integration, we split the airfoil surface into line elements from one static pressure port location to other. Element 1 is from port 1 to 2, element 2 is from port 2 to 3 and so on. Approximate the pressure on any element as the pressure on the leading port of the element, for example, the pressure everywhere on element 2 as the pressure on port 2 and so on.

From the geometry of NACA0015 we know the orientation/angle of the element relative to chord. This, along with the consideration of angle of attack, gives you the orientation of the element relative to free-stream. We then calculate the component of the pressure force (per unit span, it is length of the element times the pressure on the element) perpendicular to

Port no.	1	2	3	4	5	6	7	8	9	10	end
d_i(mm)	0	3	5	7	9	22	29	36	43	50	65
x/c	0	0.046	0.077	0.108	0.138	0.338	0.446	0.554	0.662	0.769	1
y/c	0	0.043	0.053	0.060	0.065	0.075	0.070	0.062	0.050	0.037	0.016
-y/c	0	-0.043	-0.053	-0.060	-0.065	-0.075	-0.070	-0.062	-0.050	-0.037	-0.016

Table 3: Position of Pressure Probes on the Airfoil Section

free-stream and add the vertical component forces on all elements. This gives us lift per unit span. We calculate the coefficient of lift from that.

There are 9 panels between 10 ports. The inclination θ_i of each panel can be calculated using

$$tan(\theta_i) = \frac{y_{i+1} - y_i}{x_{i+1} - x_i} = \frac{(y/c)_{i+1} - (y/c)_i}{(x/c)_{i+1} - (x/c)_i}$$

Panel no.	1	2	3	4	5	6	7	8	9	10
Ports	1-2	2-3	3-4	4-5	5-6	6-7	7-8	8-9	9-10	10-end
tan(θ_i)	0.930	0.328	0.230	0.167	0.047	-0.043	-0.078	-0.104	-0.124	-0.153
$\theta_i{}^\circ$	42.93	18.15	12.94	9.52	2.68	-2.46	-4.49	-5.94	-7.08	-8.72

Table 4: Inclination of Upper Surface Panels relative to the airfoil chord

Panel no.	1	2	3	4	5	6	7	8	9	10
Ports	1-2	2-3	3-4	4-5	5-6	6-7	7-8	8-9	9-10	10-end
tan(θ_i)	-0.930	-0.328	-0.230	-0.167	-0.047	0.043	0.078	0.104	0.124	0.153
$\theta_i{}^\circ$	-42.93	-18.15	-12.94	-9.52	-2.68	2.46	4.49	5.94	7.08	8.72

Table 5: Inclination of Lower Surface Panels relative to the airfoil chord

Furthermore, we have taken measurements for an angle of attack (α) = 2°. Thus the inclination of the panel relative to the free-stream can be given by

$$\theta_{free-stream_i} = \theta_i - \alpha$$

Panel no.	1	2	3	4	5	6	7	8	9	10
Ports	1-2	2-3	3-4	4-5	5-6	6-7	7-8	8-9	9-10	10-end
$\theta_{free-stream-upper}{}^\circ$	40.93	16.15	10.94	7.52	0.68	-4.46	-6.49	-7.94	-9.08	-10.72
$\theta_{free-stream-lower}{}^\circ$	-44.93	-20.15	-14.94	-14.52	-4.68	0.46	2.49	3.94	5.08	6.72

Table 6: Inclination of Upper and Lower Panels relative to the free-stream

To find the lift, we add the contribution of pressure force in the vertical direction from the the lower surface and subtract the contribution of pressure force from the upper surface. For this, we find the length s_i of each panel element first. We find the Pressure Force Contribution (dF).

$$dF = p \times dx = p \times s.cos(\theta)$$

Panel no.	1	2	3	4	5	6	7	8	9	10
Ports	1-2	2-3	3-4	4-5	5-6	6-7	7-8	8-9	9-10	10-end
s_i (mm)	4.097	2.105	2.052	2.028	13.014	7.006	7.022	7.038	7.053	15.175

Table 7: Length of Panel Elements

Converting panel length to metres, and calculating Lift per unit span (N/m) as

$$Net\ Force\ per\ unit\ length = \sum F_l - \sum F_u$$

Thus, we get $\boxed{Lift\ per\ unit\ span = \mathbf{1.0587\ N/m}}$

We can now calculate C_L as

$$C_L = \frac{Lift/span}{0.5\rho_\infty u_\infty^2 c}$$

Thus, we get $\boxed{C_L = \mathbf{0.256}}$

Inference

We can see that the distribution of pressure over the airfoil at a positive angle of attack generates lift in a symmetric NACA0015 airfoil.