57:020 Mechanics of Fluids and Transfer Processes
Laboratory Experiment #3

Measurement of Pressure Distribution and Lift for an Airfoil

Purpose

The objectives of the experiment are to examine the surface pressure distribution and to compute the lift force acting on the airfoil.

Test Design

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. The geometric and dynamic characteristics of airfoils are shown in Figure 1. This experiment is concerned with computation of the lift on a stationary airfoil mounted in the test section of a wind tunnel. We will consider only two-dimensional airfoils where tip and root effects are neglected.

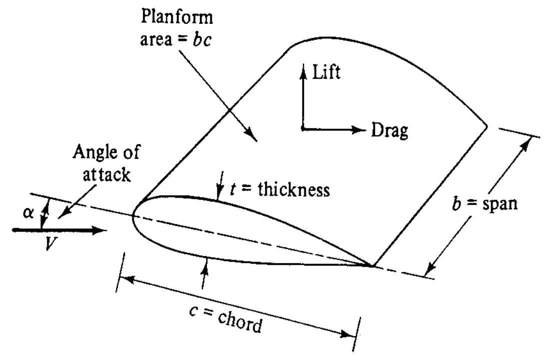


Figure 1. Geometric and dynamic parameters of airfoils

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.

There are a variety of ways to measure 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. Typical pressure distribution on an airfoil and its projection on the airfoil normal are shown in Figure 2.

The pressure distribution on the airfoil is expressed in dimensionless form by the pressure coefficient C_p

$$C_p = \frac{p_i - p_\infty}{\frac{1}{2}\rho U_\infty^2} \tag{1}$$

where p_i is the surface pressure measured at location i on the surface, p_∞ is the pressure in the free stream, ρ is air density, and U_∞ is the free-stream velocity given by

$$U_\infty = \sqrt{\frac{2\left(p_{stagnation} - p_\infty\right)}{\rho}} \tag{2}$$

where $p_{stagnation}$ is the stagnation pressure measured at the tip of the Pitot tube.

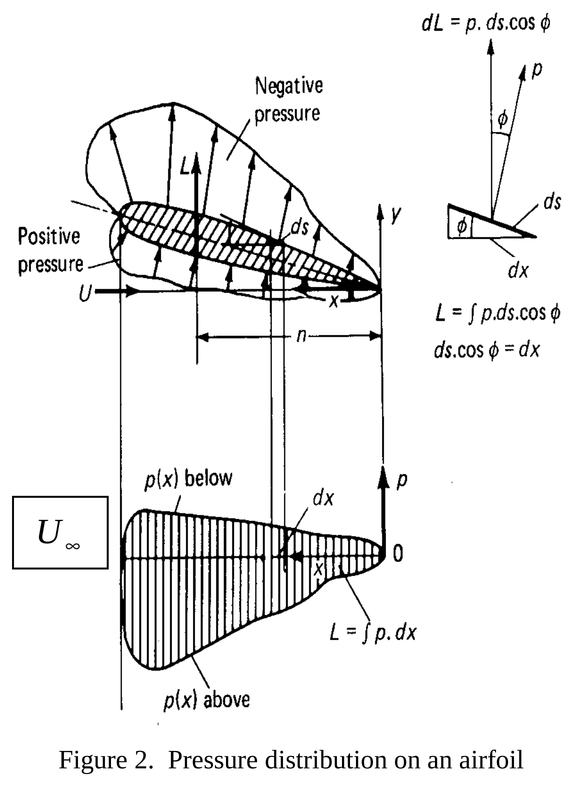


Figure 2. Pressure distribution on an airfoil

The lift force is customarily expressed as a dimensionless lift coefficient per unit span length

$$C_L = \frac{2L}{\rho U_\infty^2 bc} \tag{3}$$

where L is lift force on the airfoil obtained by integration of the measured pressure distribution over the airfoil's surface, b is the airfoil span, and c is the airfoil chord, as shown in Figure.

In this experiment, a Clark-Y airfoil is the shape under consideration. Airfoil characteristics are provided in Appendix A. Measurements will be made using an automated data acquisition system (ADAS) as sketched in Figure 3.

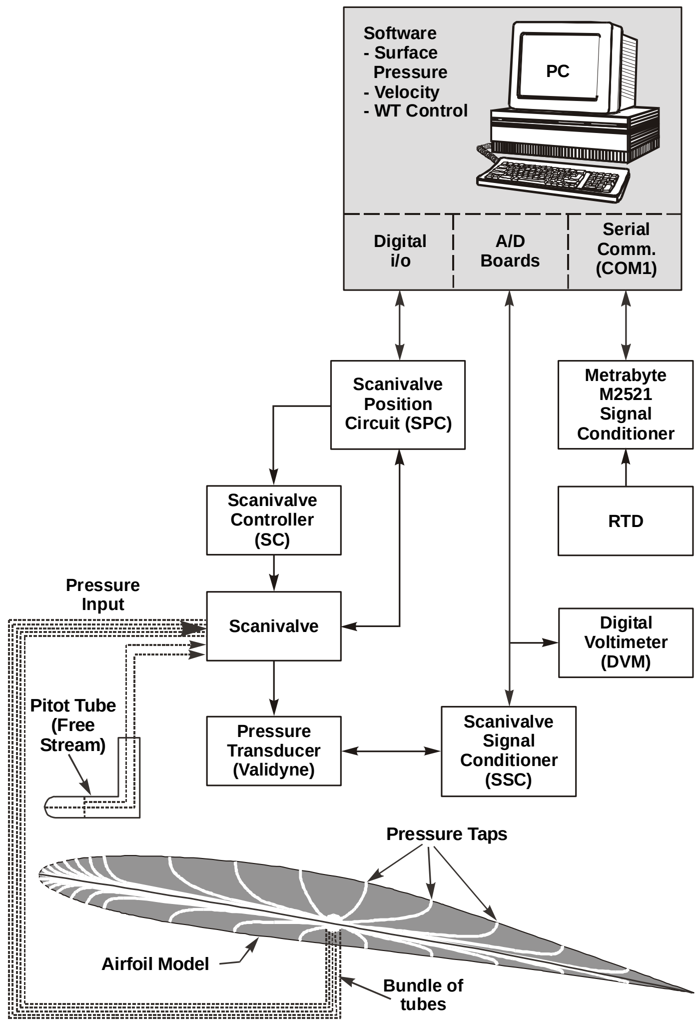


Figure 3. Schematic of the data acquisition system

Substituting in equation (3) for the measured variables and using the notations provided in Appendix A, the data reduction equation for the lift coefficient is

$$C_L = \frac{\int_s (p_i - p_\infty)\sin(\theta)\,ds}{1/2\,\rho U_\infty^2 c} \tag{4}$$

where θ is the angle of surface normal to free-stream flow.

Measurement Systems and Procedures

The equipment (measurement systems) used in this experiment includes:

- Protractor - angle of attack
- Measuring tape - chord length
- Resistance temperature detectors (RTD)/electronic data acquisition - temperature in the tunnel
- Pitot-static probe/scanning valve/pressure transducer/electronic data acquisition - free-stream velocity
- Pressure taps/scanning valve/ pressure transducer/electronic data acquisition - surface pressure

Pressure and temperature are measured using the ADAS. The pressure differences $p_i - p_\infty$ and $p_{stagnation} - p_\infty$ needed in equations (2) and (3) are measured using a pressure transducer, i.e., a device which converts a pressure difference into a voltage. The pressure transducer has one of the ports connected at all times to p_∞, the reference pressure in the free stream. The other port is connected to the output of an analog multiplexer (scanning valve), which measures pressure at various locations in test section. The scanning valve is a rotary multiple-port switch that permits multiple pressure inputs to be connected to a common output. In the present experiment the input connections to the scanning valve are as follows:

- Ports 0 to 28 are connected to the corresponding pressure taps located on the airfoil (see Figure 3)
- Port 40 is connected to the Pitot-static tube to measure the stagnation pressure
- Port 42 is connected to the Pitot-static tube to measure the static pressure (not used in experiments).

Prior to the experiments, the pressure transducer is calibrated against a Rouse manometer (precision micro-manometer with a resolution of 0.001 inches). The calibration (conversion) equation valid for the entire range of measured pressures is given by

$$h = aV + b \tag{5}$$

where h is the reading of the Rouse manometer (in inches of liquid column), V is the voltage produced by the pressure transducer, a and b are coefficients obtained through linear regression applied to the calibration points.

Figure 4 illustrates the block diagram for the measurement systems and data reduction equations for determination of the lift coefficient.

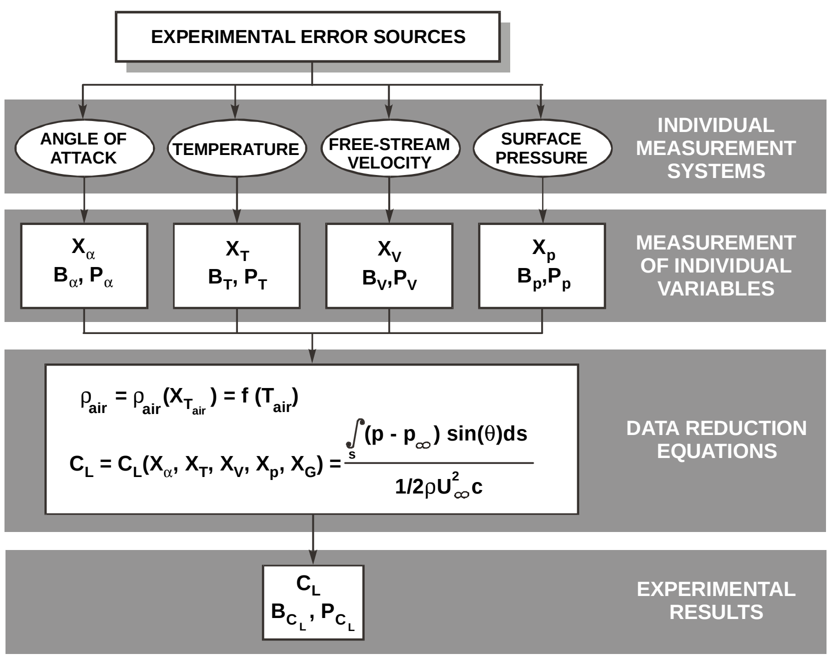


Figure 4. Block diagram of the lift coefficient determination including measurements systems, data reduction equations, and results

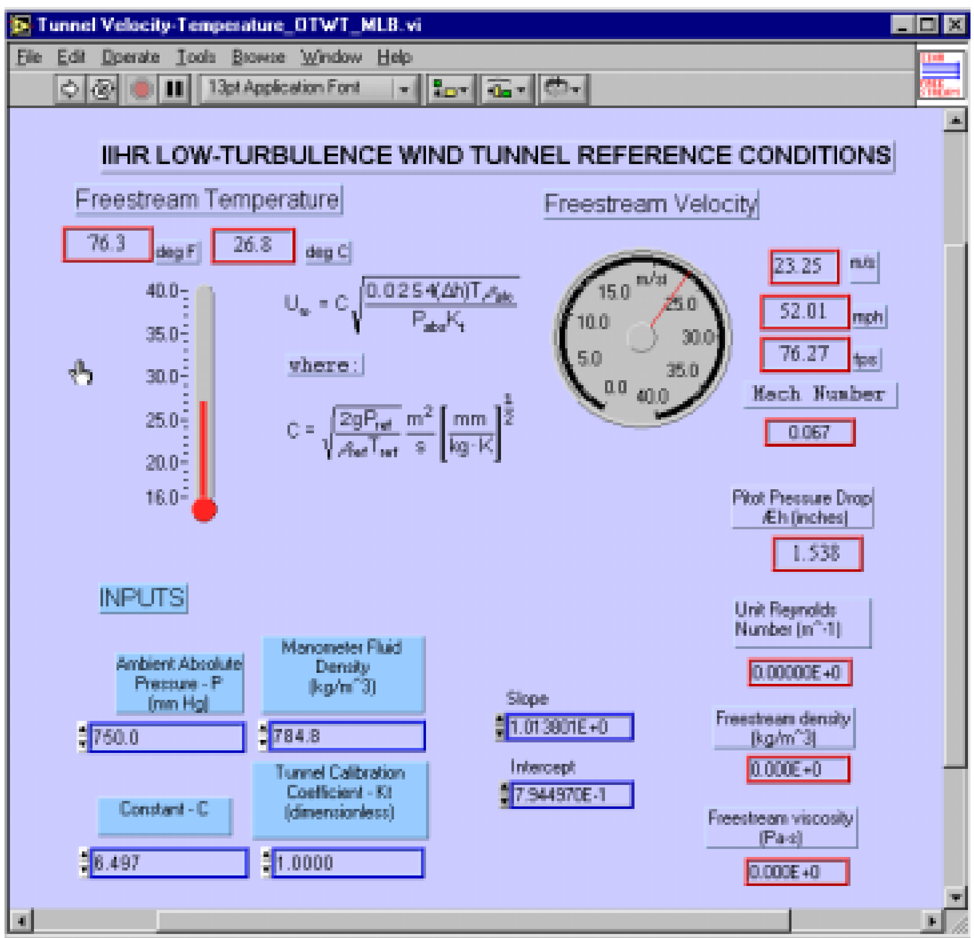
Each group of students will measure the pressure distribution on the airfoil set at an angle of attack in a pre-established value for the free-stream velocity. Each group will repeat the measurements ten times to provide information for estimation of the precision errors.

The following operations will be made during the experiment:

(1) Follow teaching assistant (TA) instructions for setting wind tunnel velocity.

(2) Follow teaching assistant (TA) instructions for powering on the wind tunnel and ADAS.

(3) Open the Fluids lab folder located on the ADAS computer monitor. Double click on the icon Tunnel Velocity – Temperature_OTWT_MLB.vi (see Figure 5). This program is a Virtual Instrument (VI) created using LabView for automatic display of the operating parameters of the wind tunnel. Enter the values of the atmospheric pressure and ambient temperature. These values can be obtained using the barometer-temperature device installed inside the large wind tunnel. Run the VI by clicking on the arrow button to the top left of the screen. The VI displays the velocity and temperature in the test section of the tunnel. Next select *Operate/Make Current Values Default* from the Menu toolbar to transmit this information to all the running VIs.



(4) In the fluids lab folder you will also find the *Lift* VI. Double click on the icon labeled *Lift*. This Virtual Instrument (VI) allows automatic acquisition of the data required for calculating the lift coefficient. The screen of the VI is shown in Figure 6. Enter in the *Observation Point List* box the scanning valve port numbers where you would like to take measurements (see above the labeling for the input connections to the scanning valve). Use commas or spaces between the port numbers in the *Observation Point List* box. Follow TAs instructions for appropriately selecting the *Sampling Rate* (Hz), *Settling Time* (s), and *Length of each Sample* (s) and enter these values in the corresponding boxes. Type in the conversion (regression) coefficients *a* and *b* (usually displayed on the computer cabinet or provided by TAs) the density of the Rouse manometer (provided by TAs), and the angle of attack at which the model is set.

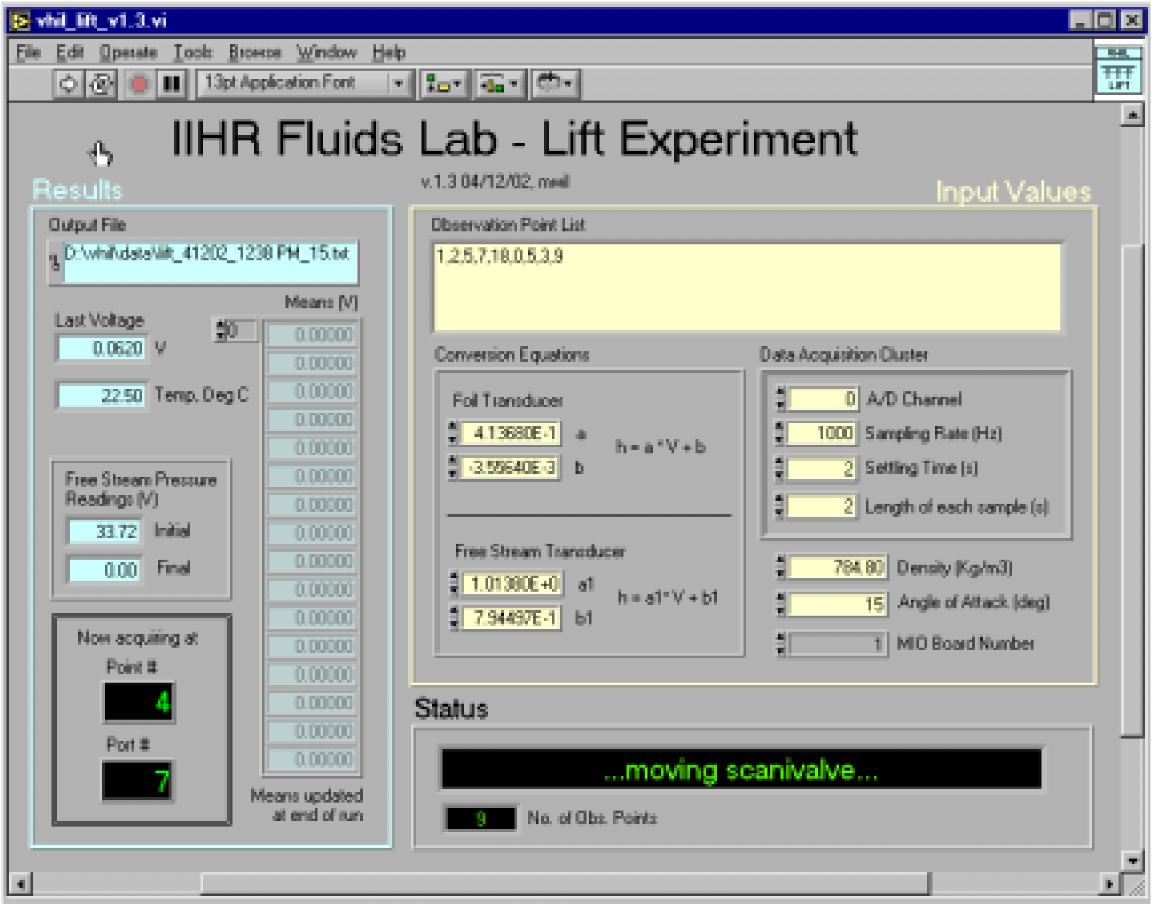


Figure 6. Computer interface for the lift measurements VI

(5) Run the VI by clicking on the arrow button to the top left of the screen. A window will appear asking about the directory where the data file will be saved. Select the file name and path of the directory to save the data.
(6) Repeat steps 4 and 5 for the remaining measurements.

Uncertainty Analysis

Using your own data, derive and present results for uncertainties estimates for pressure coefficient and lift coefficient. The methodology for estimating uncertainties follows the procedures presented in IIHR Report No. 406 (Stern et al., 1999) for Multiple Tests method (sections 6.1 and 6.3). Based on previous experiments, it was found that bias and precision limits for the ρ, U_∞, θ, s, and c are negligible, hence for the present analysis only include the bias limit for $p_i - p_\infty$. Table 5 provides bias limit estimates for the individual variables. The data reduction equations for the pressure and lift coefficients are equations (1) and (4), respectively.

Table 1. Assessment of elemental errors for the individual variables included in the data reduction equations

Variable	Bias Limit	Estimation	Precision Limit	Estimation
p_i-p_∞ **(Pa)**	2.1	Previous meas.	Determined	Repeated measurements
ρ **(kg/m^3)**	0	Negligible	0	No repeated meas.
U_∞ **(m/s)**	0	Negligible	0	No repeated meas.
$\theta = (\beta - \alpha)$ **(degrees)**	0	Negligible	0	No repeated meas.
s **(m)**	0	Negligible	0	No repeated meas.
c **(m)**	0	Negligible	0	No repeated meas.

Pressure Coefficient

The data reduction equation for the pressure coefficients (equation 1) is of the form

$$C_p = f(p_i - p_\infty, \rho, U_\infty)$$

However, here we will only consider bias limits for $p_i - p_\infty$ (see Table 1).

The total uncertainty for the pressure coefficient measured **at each pressure tap** is given by equation (24) in Stern et al. (1999)

$$U_{Cp}^2 = B_{Cp}^2 + P_{Cp}^2 \qquad (6)$$

Bias Limits. The bias limit for each of the taps (constant for all) in equation (6) is given by equation (14) in Stern et al. (1999). Neglecting the correlated bias limits and discarding the terms with negligible bias limit

$$B_{Cp}^2 = \sum_{i=1}^{j} \theta_i^2 B_i^2 = \theta_{(pi-p\infty)}^2 B_{(pi-p\infty)}^2 \qquad (7)$$

where B_i is the bias limit for the individual variable and $\theta_i = \partial C_p / \partial X_i$ are sensitivity coefficients. The sensitivity coefficient $\theta_{(p_i - p_\infty)}$, evaluated using the average values for the individual variables, is given by

$$\theta_{(p_i - p_\infty)} = \frac{\partial C_p}{\partial (p_i - p_\infty)} = \frac{2}{\rho U_\infty^2}$$

Precision limits. The precision limit for each pressure tap in equation (6) is estimated using equation (23) in Stern et al. (1999) with $t = 2$

$$P_{Cp} = 2 S_{Cp} / \sqrt{M} \qquad (8)$$

where S_{Cp} is the standard deviation of the pressure coefficients at each pressure tap evaluated using equation (22) in Stern et al. (1999) for the $M = 10$ repeated tests. Precision limits will have different values for each of the 28 taps.

Lift Coefficient

The data reduction equation for the lift coefficient (equation 4) is of the form

$$C_p = f(p_i - p_\infty, \theta, s, \rho, U_\infty, c)$$

However, here we will only consider bias limits for $p_i - p_\infty$ (see Table 1).

The total uncertainty for the measurement of the lift coefficient is given by (equation 24 in Stern et al., 1999)

$$U_{CL}^2 = B_{CL}^2 + P_{CL}^2 \qquad (9)$$

Bias Limits. The bias limit in equation (9) is given by equation (14) in Stern et al. (1999). Neglecting the correlated bias limits and the terms assumed with negligible bias limits

$$B_{CL}^2 = \sum_{i=1}^{j} \theta_i^2 B_i^2 = \theta_{(pi-p\infty)}^2 B_{(pi-p\infty)}^2 \qquad (10)$$

Given the above assumptions and replacing the integral in the data reduction equation (4) with summation of elemental lift coefficients over the surface of the airfoil, the expression for the bias limit of the lift coefficient is

$$B_{CL}^2 = B_{(pi-p\infty)}^2 \sum_{i=1}^{k} \left[\frac{\partial C_L}{\partial (p_i - p_\infty)} \right]^2 = \left[\frac{B_{(pi-p\infty)}}{0.5 \rho U_\infty^2 c} \right]^2 \sum_{i=1}^{k} [\sin(\theta_i) ds_i]^2 \qquad (11)$$

Notations used in equation (11) are defined in Table 1 and Appendix A ($k = 29$).

Precision limits. The precision limit in equation (10) is estimated using equation (23) in Stern et al. (1999) with $t = 2$

$$P_{CL} = 2S_{CL} \Big/ \sqrt{M} \qquad (12)$$

where S_{CL} is the standard deviation of the lift coefficients evaluated using equation (22) in Stern et al. (1999) for the $M = 10$ repeated tests.

Data Analysis and Discussions

Each group of students will obtain pressure coefficients and lift coefficients for one free-stream velocity and an angle of attack. All data required for analysis is contained in a file outputted by the ADAS. The format of the outputted file is illustrated in Figure 7

	A	B	C	D	E	F	G
1	'Legend'						
2	**'File Name'**						
3	' *Angle (degrees), Temperature (°C), Manometer Fluid Density (kg/m³)*'						
4	'Coefficients *a*,*b*, from *h* = *a*·*V* +*b* conversion equation						
5	'Coefficients a1, b1 from h = a1 * V + b1 conversion equation						
6	'***A/D channel, Sampling rate (Hz), Settling Time (s), Sampling time (s)***'						
7	'===================================='						
8	**D;\vhil\data\lift_041202_0911_15.txt**						
9	10	25.1	784.8				
10	4.14E-01	-3.56E-03					
11	1.01E+00	7.94E-01					
12	0	1000	2	2			
13	'===================================='						
14	'Data Section – Pressure Tap Number (Scanivalve Port), Mean Voltage (V)'						
15	0	-0.0703					
16	1	-0.66949					
17	2	-0.78618					
18	Etc.	Etc.					
19	27	0.38178					
20	28	0.5081					
21	40	0.53052					

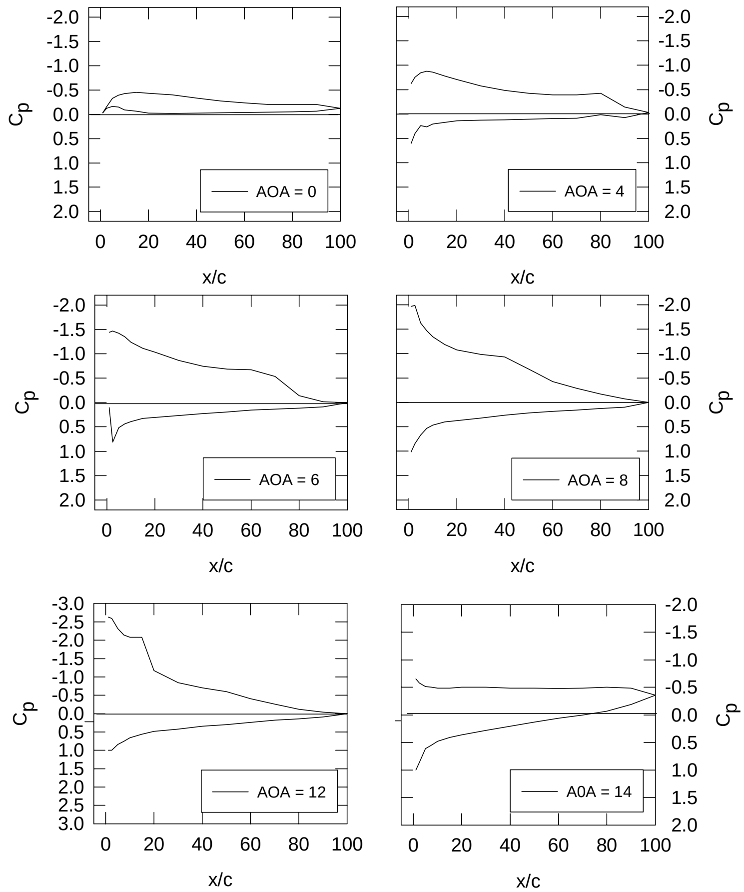
Figure 7. Sample of data file outputted by ADAS

The values for the items/quantities listed in rows 2 to 6 are specified in the same order in rows 8 to 12. Quantities listed from row 15 on are those measured by ADAS. Values in column A specify the port where the differential pressure is measured. Values in column B are voltages measured by the pressure transducer. Using the conversion equation (5) these values can be converted into inches of manometer liquid (i.e., alcohol ρ = 784.8kg/m^3 at 26°C) and subsequently in pressures. Note that the density of the Rouse liquid (usually alcohol) varies with the ambient (not wind tunnel) temperature.

Data reduction will include the following steps:

1. Calculate the flow Reynolds number, $\mathrm{Re} = U_\infty \cdot c/\nu$ (using equation 3 and the reading on port 40).
2. Calculate the pressure coefficients (using equation 2 and the readings on ports 0 to 28) on the airfoil surface. Plot C_p versus x/c, for the angles of attack employed in the experiment. Include as error bars on this plot the calculated total uncertainties. Compare your distribution with the reference data provided in Figure 8.a.

a)



b)

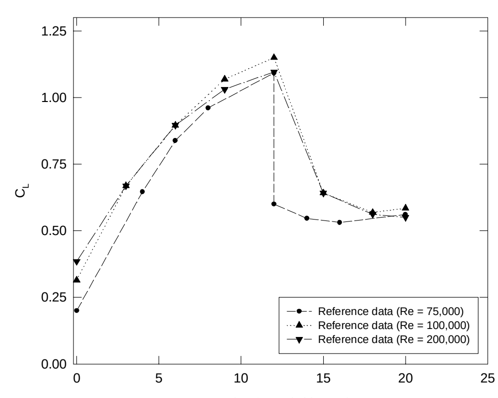


Figure 8. Reference data (Marchman and Werme, 1984); a) Distribution of the pressure coefficients for $\alpha = 0°$, 4°, 6°, 8°, 12°, 14° and Re = 75,000; b) Variation of the lift coefficient with the angle of attack

3. Calculate the experimental lift force on the airfoil. The lift force is obtained by integrating the measured pressure over the airfoil surface. Use of the trapezoidal rule for the numerical integration is recommended (see Appendix A).
4. Calculate the lift coefficient per unit span length, C_L, given by equation (4). Include the uncertainty estimate for the lift coefficient. Compare your value for the lift coefficient with the value determined from the reference data provided in Figure 8.b.

Answer the following questions:

1. Suggest another experimental procedure to measure the lift force on the airfoil.
2. If the lift L is a function of the free-stream velocity U, density ρ, chord c, angle of attack α, and viscosity μ, what are the dimensionless groups (π parameters) that characterize this problem?
3. How do the experimental measurements of the pressure distribution apply to a full-scale aircraft having a similar airfoil section?
4. How do the experimental measurements of the pressure distribution apply to a full-scale aircraft having a similar airfoil section?

References

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APPENDIX A

Geometry of the Clark-Y Airfoil and Positioning of the Pressure Taps on the Airfoil Surface

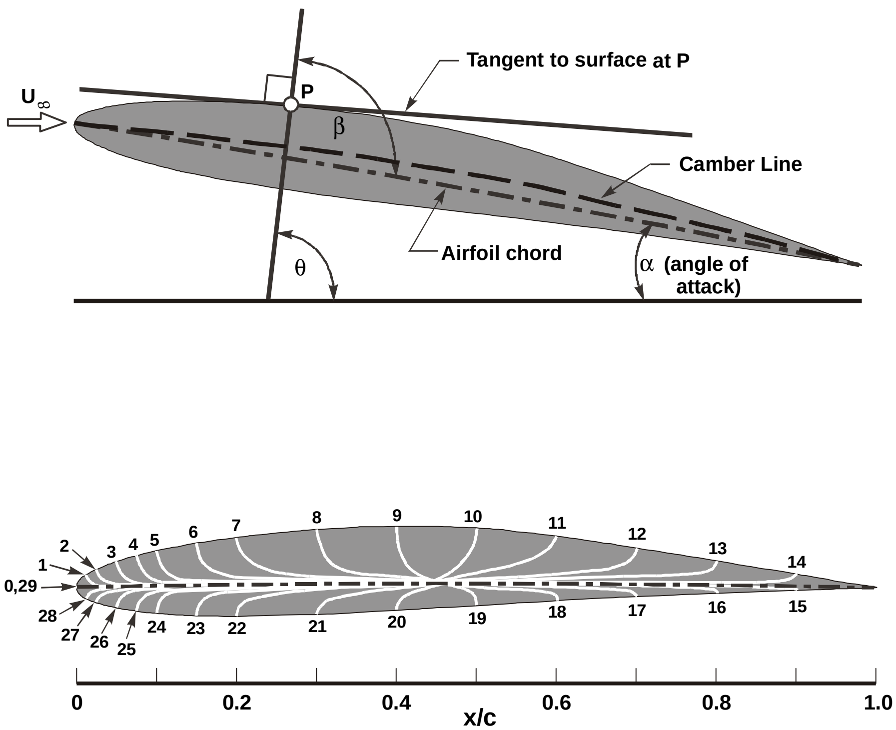

Figure 1A. Positioning of the pressure taps on the airfoil surface

N Node Number (i)	x/C	Δs Surface Element Length (mm)	β Surface (i-1) to (i) to Chord Line Angle (degrees)
0	0.00		
1	1.25	6.0	160.7
2	2.50	6.0	136.2
3	5.00	9.0	115.8
4	7.50	9.0	107.3
5	10.00	8.0	102.6
6	15.00	15.5	100.1
7	20.00	15.5	97.1
8	30.00	30.0	93.2
9	40.00	30.0	89.8
10	50.00	30.0	86.7
11	60.00	31.5	83.9
12	70.00	31.5	81.9
13	80.00	31.5	81.1
14	90.00	31.5	79.7
Trailing Edge	100.00	30.0	78.6
15	90.00	29.0	272.4
16	80.00	31.0	272.4
17	70.00	31.0	272.4
18	60.00	31.0	272.4
19	50.00	31.0	272.4
20	40.00	30.0	272.4
21	30.00	30.0	272.4
22	20.00	30.0	272.4
23	15.00	15.5	272.4
24	10.00	15.5	272.4
25	7.50	7.5	272.4
26	5.00	9.0	265.5
27	2.50	6.0	253.4
28	1.25	4.5	241.0
29 or 0	0.00	8.0	209.0

Airfoil Geometry:

- maximum thickness, $t = 0.0254$ m;
- airfoil wing span, $b = 0.762$ m;
- chord length, $c = 0.3048$ m

Specific notations:

- surface to chord line angle, β;
- angle of surface normal to free-stream flow, θ;
- angle of attack, α.

Lift force can be computed using the trapezoidal scheme for the numerical integration

$$L = \oint_s (p_\infty - p)\sin(\theta)ds \approx \sum_i \frac{1}{2}[(p_\infty - p_i) + (p_\infty - p_{i-1})]\sin(\theta_{i-1,i})\Delta s_{i-1,i}$$