AerE 344: Undergraduate Aerodynamics and Propulsion Laboratory

Lab Instructions

Lab #05: **Determination of the Aerodynamic Performance of a Low-Speed Airfoil based on Pressure Distribution Measurements**

Purpose:

- To learn how to operate a low-speed wind tunnel.
- To learn how to use Bernoulli's equation for the low-speed wind tunnel test.
- To learn how to use the wind tunnel calibration factor for low-speed flow velocity measurements.
- To learn how to determine the aerodynamic characteristics of the airfoil based on pressure measurement
- To learn how to plot and analyze aerodynamic performance curves.

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AerE344 Lab 05: Determination of the Aerodynamic Performance of a Low-Speed Airfoil based on Pressure Distribution Measurements

In this lab exercise, you will use the pressure sensors you used last week, i.e., three Scanivalve pressure transducer DSA3217 units, to acquire pressure from the pressure taps around a GA(W)- 1 airfoil to determine the aerodynamic performance of the airfoil.

What will be available to you for this lab:

- A thermometer and barometer for observing ambient lab conditions (for calculating atmospheric density).
- A computer with a data acquisition system capable of measuring the voltage from your manometer.
- A GA(W)-1 airfoil that can be mounted at any angle of attack up to 16.0 degrees.
- Three 16-channel Scanivalve DSA electronic pressure scanners.

Steps:

- Choose a primary operator and have the TA record your choice.
- Choose a wind tunnel velocity at which to conduct your tests (incoming flow velocity = 10~15m/s is recommended).
- Check all the connection are correct for the airfoil and P_E and P_A. Conduct your airfoil pressure measurement experiments.
    - Use the Scanivalve pressure transducers to measure pressures about the GA(W)-1 airfoil in the Blue Wind Tunnel.
    - Measure pressure distributions for the following angles of attack:
    - $AOA = -4°, 0°, 4°, 6°, 8°, 10°, 12°, 14°, 16°$ and/or any others you wish. You can set the angle of attack by mounting the protractor on the end plate of the airfoil model in the wind tunnel.
    - **Important:** DO NOT TOUCH THE AIRFOIL WHEN CHANGING THE ANGLE OF ATTACK. ONLY HANDLE THE BARS ON EACH END.
    - Repeat your measurements by resetting the airfoil at the above angles of attack as many time as you have time for. This will allow you to reduce uncertainty.

Calculating airfoil lift coefficient (C_L), drag coefficient (C_D), and moment (coefficient) ($C_{m,LE}$) by numerically integrating the surface pressure distribution around the airfoil:

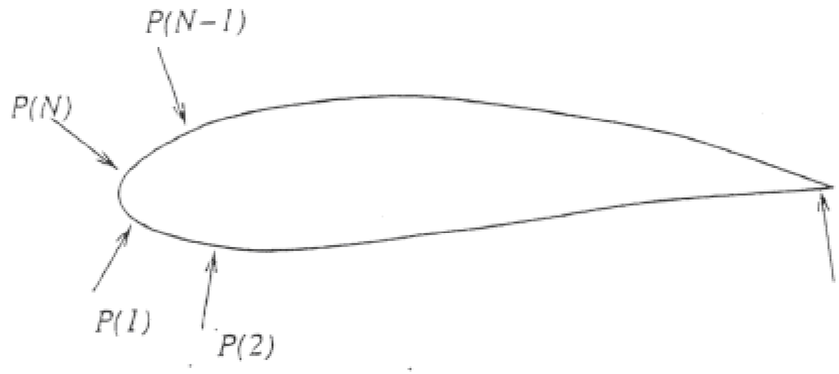

Figure 1. Pressure tap numbering convention

First, recall that the surface pressure taps are numbered in the counterclockwise direction as shown in Fig. 1. Although it may seem somewhat unintuitive at first, this numbering convention allows us to formulate relevant equations in a very generic way.

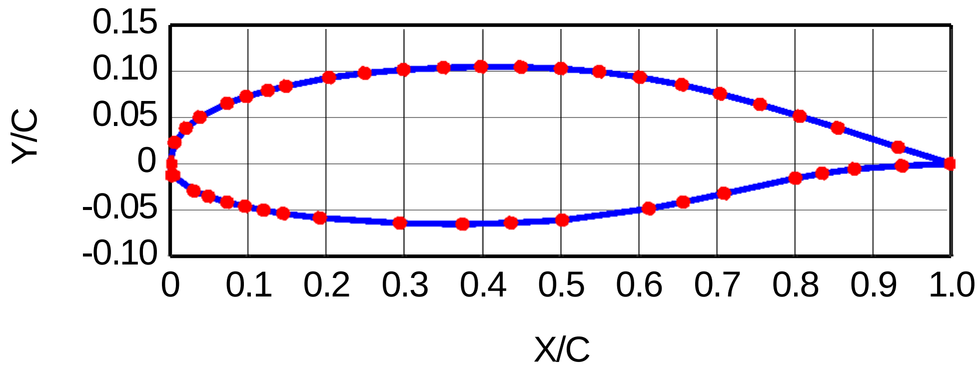

Figure 2. GA(W)-1 airfoil and pressure tap locations.

Table 1: The coordinate of the pressure taps on the GA(W)-1 airfoil.

Lower Surface			Upper Surface		
tap	x/c	y/c	tap	x/c	y/c
1	0.0000	0.0000	22	0.9321	0.0177
2	0.0036	-0.0126	23	0.8549	0.0386
3	0.0306	-0.0293	24	0.8059	0.0514
4	0.0494	-0.0355	25	0.7552	0.0639
5	0.0735	-0.0418	26	0.7042	0.0755
6	0.0962	-0.0462	27	0.6551	0.0851
7	0.1201	-0.0502	28	0.6013	0.0935
8	0.1452	-0.0539	29	0.5496	0.0992
9	0.1921	-0.0585	30	0.5003	0.1027
10	0.2944	-0.0641	31	0.4492	0.1045
11	0.3746	-0.0653	32	0.3982	0.1047
12	0.4365	-0.0640	33	0.3503	0.1036
13	0.5023	-0.0609	34	0.2992	0.1015
14	0.6130	-0.0486	35	0.2493	0.0979
15	0.6569	-0.0415	36	0.2040	0.0930
16	0.7093	-0.0322	37	0.1487	0.0838
17	0.8004	-0.0158	38	0.1256	0.0792
18	0.8348	-0.0105	39	0.0980	0.0725
19	0.8759	-0.0056	40	0.0734	0.0651
20	0.9367	-0.0023	41	0.0385	0.0503
21	1.0000	0.0000	42	0.0207	0.0383
			43	0.0063	0.0227

- TAP 1 is at the airfoil leading edge (LE), and TAP21 is at the airfoil trailing edge (TE)
- TAP 2-20 are along the lower surface; TAP 22-43 are along the upper surface
- The chord length of the airfoil is 101mm, i.e., C = 101mm.

With the convention, the airfoil surface is broken into N panels. The i-th panel is bounded by the i-th and i+1-th taps at (x_i, y_i) and (x_{i+1}, y_{i+1}) respectively. The exception is that the Nth panel is defined by (x_N, y_N) and (x_i, y_i) but in your spreadsheet or program, you can treat this by adding a fictitious N+1-th tap which takes on the value from the first tap.

Consider the i-th panel shown in Fig.2, where $p_{i+1/2}$ represents the pressure (assumed to be) acting on the i-th panel. Let

$$\begin{cases} p_{i+1/2} = \frac{1}{2}(p_i + p_{i+1}) \\ p_{N+1/2} = \frac{1}{2}(p_N + p_1) \end{cases} \tag{1}$$

If we assume the pressure variation on the i-th panel to be constant at $p_{i+1/2}$ as defined by Eq.1, this is equivalent to true trapezoidal integration. Furthermore, define

$$\begin{cases} \Delta x_i = x_{i+1} - x_i, & \Delta y_i = y_{i+1} - y_i \\ \Delta x_N = x_1 - x_N, & \Delta y_N = y_1 - y_N \end{cases} \tag{2}$$

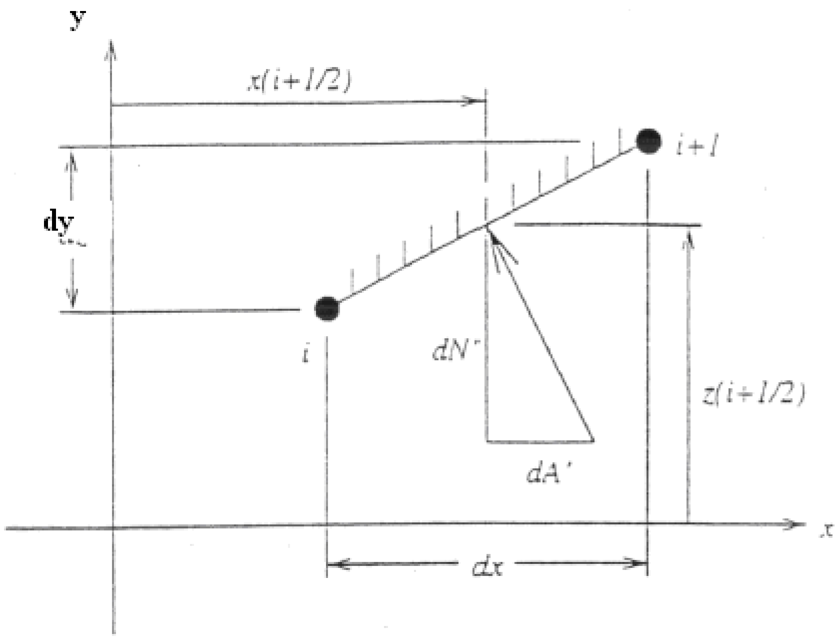

Figure 3. Discrete representation of the airfoil surface element

Note that Δx and Δy can be negative because x and y are not monotonic in the index i. Using Eqs.1 and 2, the normal and axial components of the pressure force acting on the i-th panel can be written as

$$\delta N'_i = p_{i+1/2}\Delta x_i \tag{3}$$

$$\delta A'_i = -p_{i+1/2}\Delta y_i \tag{4}$$

where the prime indicates a two-dimensional quantity (force per unit span). Similarly, the moment contribution from the i-th panel to the total moment about the leading edge can be written as

$$\delta M'_{LE,i} = \mathbf{r} \times \delta \mathbf{F}_i$$

$$= (x_{i+1/2}\mathbf{i} + y_{i+1/2}\mathbf{k}) \times (\delta A'_i \mathbf{i} + \delta N'_i \mathbf{k})$$

$$= (x_{i+1/2}\mathbf{i} + y_{i+1/2}\mathbf{k}) \times (-p_{i+1/2}\Delta \mathrm{z}_i \mathbf{i} + p_{i+1/2}\Delta \mathrm{x}_i \mathbf{k})$$

$$= -[(p_{i+1/2}\Delta x_i)x_{i+1/2} + (p_{1+1/2}\Delta y_i)y_{i+1/2}]\mathbf{j} \qquad (5)$$

or

$$\delta M'_{LE,i} = -(p_{i+1/2}\Delta x_i)x_{i+1/2} - (p_{1+1/2}\Delta y_i)y_{i+1/2}$$

where

$$x_{i+1/2} = \frac{1}{2}(x_i + x_{i+1}), \qquad y_{i+1/2} = \frac{1}{2}(y_i + y_{i+1})$$

Note that the aerodynamic moment is defined to be positive in the pitch-up direction. Now that we have derived the expressions for the differential force and moment from each panel, we can integrate them over the airfoil surface.

$$N' = \sum_{i=1}^{N} \delta N'_i = \sum_{i=1}^{N} p_{i+1/2}\Delta x_i \qquad (6)$$

$$A' = \sum_{i=1}^{N} \delta A'_i = -\sum_{i=1}^{N} p_{i+1/2}\Delta y_i \qquad (7)$$

$$M'_{LE} = \sum_{i=1}^{N} \delta M'_{LE}$$
$$= -\sum_{i=1}^{N}(p_{i+1/2}\Delta x_i)x_{i+1/2} - \sum_{i=1}^{N}(p_{i+1/2}\Delta y_i)y_{i+1/2} \qquad (8)$$

Finally, the lift and drag per unit span can be obtained as follows.

$$L' = N'\cos\alpha - A'\sin\alpha$$
$$D' = N'\sin\alpha + A'\cos\alpha$$

Writeup Guidelines

The report for this project should be a formal report. See the course website for the details on exactly what constitutes a formal report.

Required Plots:

- Plots of C_L, C_D, and C_M vs. angle of attack for the GA(W)-1 airfoil.
- Plot of C_P at angles of attack $AOA = -4°, 0°, 4°, 6°, 8°, 10°, 12°, 14°, 16°$ for the GA(W)-1 airfoil.

Your report must provide details on:

- The flow speed you used for the airfoil pressure distribution measurements.
- Discussion of the plots of the pressure coefficient (C_P) distributions of the airfoil.
- Discussion of C_L, C_D, and C_M calculated from C_P distributions—and how you calculated them. (Show the detail process)
- Estimates of the location of the stagnation point for each angle of attack.
- Estimate of the stall angle (if possible) from your measurements.
- Calculate tunnel velocity, and Reynolds number of tests (with respect to the airfoil chord length of 101 mm).