The University of Queensland
School of Mechanical and Mining Engineering

MECH3410 - Fluid Mechanics
Experiment 1 - Measurement of Lift and Drag on an Aerofoil
Location - Hawken Building 104

Introduction

The flow of fluid past an aerofoil or hydrofoil is of practical importance in many applications, notably fixed and rotary wing aircraft, hydrofoil craft, sailing boats, turbines, compressors and fans. The wind tunnel techniques used to measure aerodynamic forces acting on a scale model of the full sized object are equally applicable in studies of flow past other streamlined or blunt bodies such, as tall chimneys, overhead power lines, building, cooling tubes in heat exchanges and marine structures. Experimentally determined surface pressures can be used to evaluate body forces acting on complex shapes that cannot be determined using analytical models. Similarly knowledge of pressure gradients in the streamwise direction give insight into regions were separation is likely to occur.

Aims

The aims of this experiment are:

1. to measure the pressure distribution over the surface of a two-dimensional aerofoil at a number of air speeds and angles of attack and to calculate the associated lift and drag forces,

2. to compare the results with theory,

3. to study the phenomenon of stall,

4. to obtain experience in manometry (the measurement of pressure), and

5. to obtain experience in applying mathematical methods to obtain net forces acting on objects.

Theory

Each of the recommended textbooks for the subject has a section on aerofoil theory. Read one of:

a) F.M. WHITE, Fluid Mechanics (7th Edition), McGraw-Hill, 2010: Chapter 8 Section 7.

b) R.W. Fox and A.T. McDonald, Introduction to Fluid Mechanics (5th Edition), Wiley, 1998, Chapter 9 Part B.

Flow of fluid past an aerofoil gives rise to a resultant force R on the aerofoil. This force can be resolved into components N and T, which are normal and parallel to the aerofoil chord line, or components L and D (lift and drag) which are normal and parallel to the direction of the undisturbed fluid flow (free stream direction). This is shown in Fig. 1.

The forces are related by

$$L = N\cos\alpha - T\sin\alpha \tag{1}$$

$$D = N\sin\alpha + T\cos\alpha \tag{2}$$

where α is the angle of attack. Although R is strictly the sum of components due to pressures normal and tangential to the surface of the aerofoil and frictional shear stresses, it has been established that in many practical situations the 'skin friction' contribution to N and T are is small and can be neglected. Furthermore, for small angles of attack the pressure component of T can be neglected also. This allows the simplification:

$$N \approx R; \qquad T \approx 0. \tag{3}$$

Hence, if the pressure distributions acting on the top and bottom of the wing are know, and if these are resolved in the wing normal direction, the force N is obtained. Using Eqns. (1 - 3) the corresponding lift and drag force can be calculated.

Pressure distribution data obtained for an aerofoil at a certain angle of attack are usually presented in dimensionless form as plots of pressure coefficient, C_p, along the chord length of the aerofoil as shown in Fig. 2. C_p is defined as

$$C_p = \frac{P}{\frac{1}{2}\rho U^2} \tag{4}$$

These contours give insight into which parts of the aerofoil experience high and low pressure and thus how they contribute to overall lift generation. Note that to obtain curves with similar values to those seen in Fig. 2, C_p must be computed from the gauge pressure (absolute pressure minus the freestream static pressure). Using absolute or gauge pressure in calculating C_p does not affect the integrated forces. Integration of the C_p curve between $x = 0$ and $x = c$ yields the normal force, N acting in direction perpendicular to the chord line. Resolving this in the direction normal to the flow and using the assumption that the pressure force acting in the wing tangential direction, T is small, allows the lift and drag force to be calculated.

Similar to pressure, lift and drag are turned into the non-dimensional lift and drag coefficients

$$C_L = \frac{L}{\frac{1}{2}\rho U^2 c S}; \qquad C_D = \frac{D}{\frac{1}{2}\rho U^2 c S} \tag{5}$$

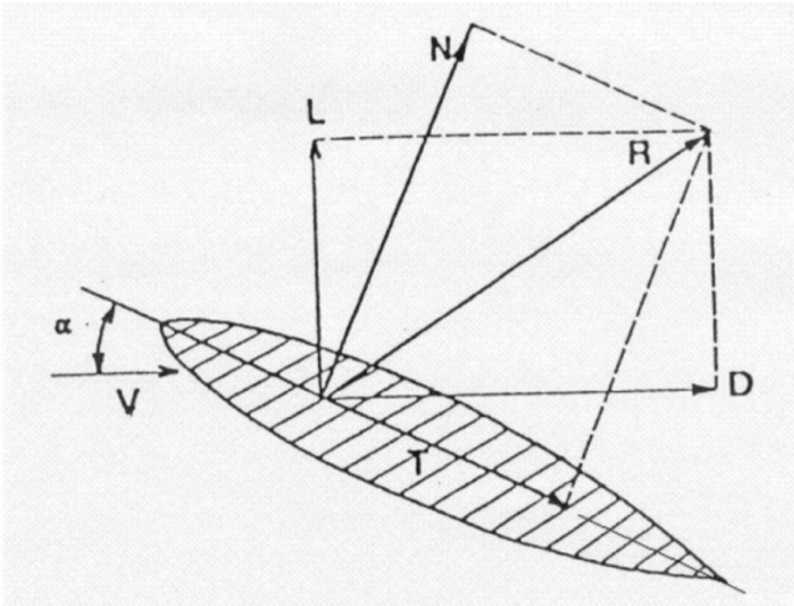

Figure 1: Components N and T or L and D of the resultant force R on the aerofoil.

where $c\,S$ is the wing planform ares, defined by chord length, c and wing span, S for two-dimensional aerofoils. Using the coefficients for pressure, lift and drag, normalised by the dynamic pressure $(\frac{1}{2}\,\rho\,U^2)$ and characteristic area, allows pressures and forces to be calculated for geometrically similar wings operating at different conditions.

Aerofoils redirect the incoming flow resulting in a pressure differences between the top and bottom of the wing. When using finite length wings, this results in the higher pressure air at the bottom surface wanting to move to the top surface at the wing tips. This phenomenon creates vortices and induces additional drag known as lift induced drag. For finite length wings the total drag is written as

$$C_D = \underbrace{C_{D,0}}_{\text{Skin Friction + Form}} + \underbrace{\frac{C_L^2}{\pi\,e\,AR}}_{\text{Lift Induced}} \tag{6}$$

In the above equation e is the aerofoil efficiency factor, which for ideal aerofoils is taken to be 1 (use this value for the current prac) but typical values fall between 0.85-0.95 and AR is the aspect ratio, which is the ratio of the span squared and aerofoil planform area,

$$AR = \frac{S^2}{A} = \underbrace{\frac{S}{c}}_{\text{rectangular aerofoils}} \tag{7}$$

In the simplest theory of flow past a thin aerofoil, viscous effects are neglected and it can be shown, after detailed analysis and application of the Kutta Condition (see, for example,

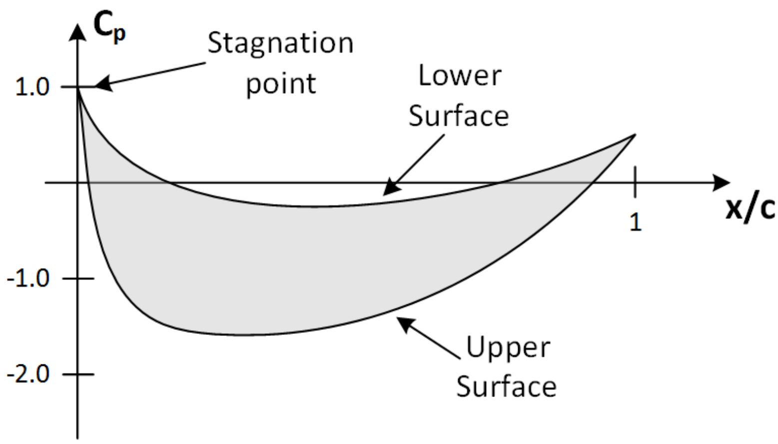

Figure 2: Pressure distribution (normalised) plotted vs wing chord length.

White, Section 8.7), that for small angles of attack

$$C_L = 2\,\pi\,\alpha \tag{8}$$

Note viscous effects will reduce the lift coefficient below this value in practice.

The dimensionless lift coefficient measured on a model at small angles of attack can usually be used with reasonable accuracy for a full scale aerofoil. This is the case as lift is predominantly generated by the pressure acting in the surface normal direction, which is primarily a function of the actual wing shape and only weakly related to viscous effects ($chord \gg \delta^*$). However this is not always true for the drag coefficient. Drag is related to Reynolds number based on the chord length, Re_c, which is defined as

$$Re = \frac{\rho\,U\,c}{\mu} \tag{9}$$

where μ is the viscosity of the fluid. This is the case as Re increases both with dimension, c and air velocity, U. Thus a full scale prototype tends to operate at much increased values of Reynolds number, implying that the flow is more turbulent in full scale applications compared to sub scale experiments. Thus, ideally sub-scale wings are tested at higher velocities, so that Reynolds number can be matched between the the model and final prototype. However often this is not possible as this requires flows Mach number > 0.3, which creates introduces compressibility effects.

Facility

The facility comprises a low speed wind tunnel as shown in Fig. 3. The tunnel consists of a suction fan located downstream of the test section, which draws air from atmospheric conditions through a test section. Some of the key design features of the tunnel are marked

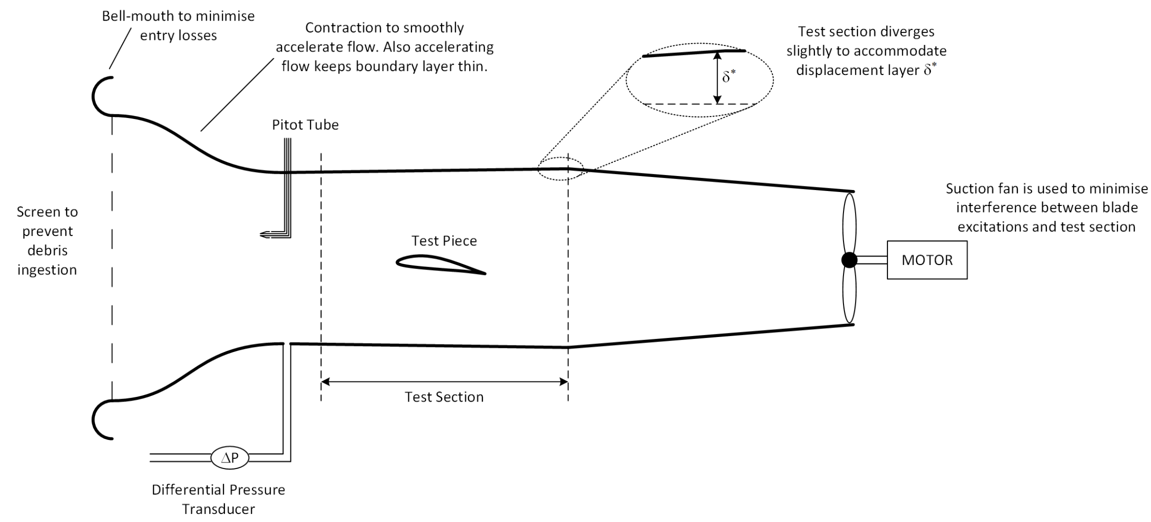

Figure 3: Schematic of the wind tunnel used for the practical.

in Fig. 3. Using the contraction a near uniform flow is generated at the start of the test section. Flow velocity in the tunnel is controlled by setting the rotational speed of the fan, which is related to the pressure just upstream of the fan. For velocity measurements the tunnel has the option to use a Pitot tube, as shown in Fig. 5 (currently not installed) or a ring of static pressure tappings mounted immediately upstream of the test section. The pressure measurements from these devices can be used to calculate air speed (see Pre-work).

Within the test section a CLARK Y 14% aerofoil is mounted vertically on a rotating

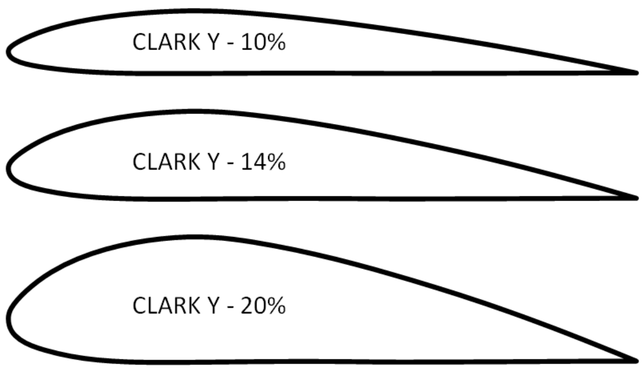

Figure 4: Profiles of Aerofoils from the Clark Y family

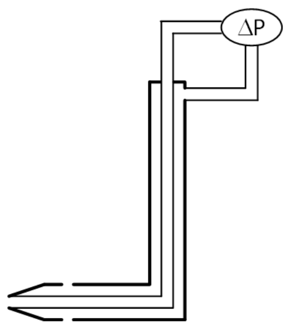

Figure 5: Pitot Tube used for measuring air speed

plate. This allows the testing at different angles of attack, α. The CLARK Y aerofoil is a common aerofoil used in general aviation. The second number (e.g. 14%) indicates the

Tapping number	Distance from leading edge, x (mm)	x/c
1	0.0	0.000
2	4.0	0.044
3	8.0	0.089
4	16.0	0.178
5, 14	25.0	0.278
6, 15	34.0	0.378
7, 16	43.0	0.478
8, 17	52.0	0.578
9, 18	61.0	0.678
10, 19	70.0	0.778
11	5.0	0.056
12	9.0	0.100
13	17.0	0.189

Table 1: Locations of pressure tappings on the CLARK Y 15 % aerofoil.

aerofoil thickness which can be scaled for a given application as shown in Fig. 4.

The aerofoil is equipped with surface pressure tappings that allow the surface pressure to be measured at different positions over the aerofoil length. Tufts on the aerofoil can be used to observe flow close to the wing surface. These tufts indicate the local flow direction. For example if the tufts point away from the wing surface this indicates that the flow direction is no longer parallel to the wing, suggesting that separation has occurred. This is an early indicator of the stall phenomena.

A manometer bank and a electronic differential pressure sensors are available to measure pressures. The locations of the pressure tappings on the aerofoil are presented in terms of distance from the leading edge in Fig. 8 and Tab. 1. Note locations are normalized by the chord length, $c = 90\,\mathrm{mm}$.

Pre-work

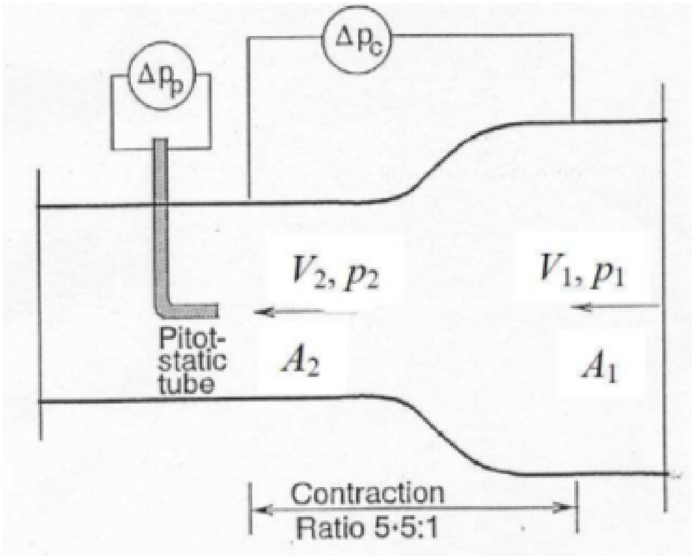

Figure 6: Schematic of test section contraction and tunnel pressure tapping locations.

Question 1: Figure 5 shows the schematic of a Pitot tube. If the measured pressure difference is given as Δp_p, show by applying Bernoulli's equation that

$$\Delta p_p = \frac{1}{2}\rho U^2 \tag{10}$$

$$\Delta p_p = P_0 - p_2, \tag{11}$$

where P_0 is the total (stagnation) pressure and p_2 is the static pressure in the test section.

Next by considering a contraction as shown in Fig. 6 and the measured pressure differentials across the contraction, $\Delta p_c = p_1 - p_2$ and using Bernoulli's equation across the contraction show that

$$\Delta p_c = \Delta p_p \left[1 - \left(\frac{A_2}{A_1}\right)^2\right] \tag{12}$$

where A_2 is the cross-sectional area of the test section and A_1 is the cross sectional area upstream of the contraction, i.e. A_2/A_1 is the area ratio across the contraction.

Finally, find an expression that relates the electronic pressure measurement ΔP (see Fig. 3), measuring the difference between static pressure in the test section and atmospheric conditions, and flow velocity in the test section.

Question 2: A Boeing 727-200 jet transport has a wing platform area $A_p = 149\,\mathrm{m}^2$

and aspect ratio $AR = 12$. The gross weight of the aircraft is 667 000 N. Below $M = 0.6$, drag due to compressibility effects is negligible, so the following equation may be used to estimate the total drag of the aircraft:

$$C_D = C_{D,0} + \frac{{C_L}^2}{\pi \, AR} \tag{13}$$

where $C_{D,0}$ is the drag at zero lift and is constant at 0.0182. What is the total drag force acting on the plane when it is flying at velocity $U_\infty = 490$ km/h at sea level?

Question 3: Experiments have been performed on a cylinder with a radius of 0.05 m in a wind tunnel, with pressure measurements made at several angles, α, as show in Fig. 7. The corresponding absolute pressure readings are given in Tab. 2. The test was performed with an inflow velocity of 25 $\mathrm{m\,s^{-1}}$ and a static pressure measured at the windtunnel wall of 85 kPa and the temperature is 25°C. This question is to give you experience in the procedure required to calculate the forces on the airfoil in the experiment.

This question is best answered using a spreadsheet (e.g. excel) or python, which can be used to plot the angular pressure variation and perform the numerical integration. Use the spreadsheet or python to do the following:

- Use the pressure measurements to calculate and plot the pressure coefficient C_p as a function of angle, α.
- By considering a segment $d\alpha = 10°$ (e.g. a segment that extends from one measurment location to the next) on the surface of a cylinder with radius R_0, draw a diagram to show how the pressure contributes to a surface force, dF.
- Resolve the force dF into horizontal and vertical components.
- Use numerical integration (e.g. the trapezoidal rule from MECH2700) to obtain the net lift and drag forces acting on the cylinder.

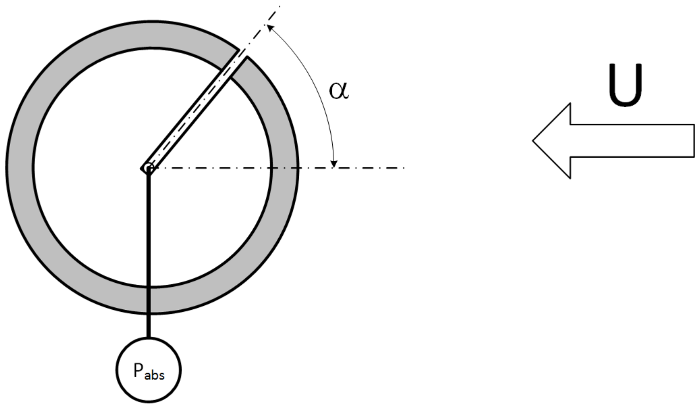

Figure 7: Schematic of test arrangement used to measure surface pressure acting on a cylinder.

Table 2: Experimental pressure data for a cylinder in the wind tunnel. (Due to symmetry, measuremens in region $0 \rightarrow -180$ are equal)

Angle (degree)	Pressure (kPa)	Angle (degree)	Pressure (kPa)
0	100.0625		
10	100.0540	100	99.9360
20	100.0305	110	99.9370
30	99.9600	120	99.9315
40	99.9565	130	99.9350
50	99.9470	140	99.9435
60	99.9370	150	99.9395
70	99.9405	160	99.9380
80	99.9300	170	99.9380
90	99.9360	180	99.9340

Experimental Procedure

1. The demonstrator will first discuss the general features of the wind tunnel and its operating procedure.
2. By eyes, set the aerofoil to zero angle of attack.
3. In steps of 100 rpm increase the fan speed and record the pressures from the differ-ential pressure transducer and the pitot tube. Stop when pressure reaches 500 Pa. Use the formulae developed in your pre-work to create a chart of fan speed vs velocity.
4. Run the tunnel to a stable speed of 10 m s^{-1}
5. Set the wing to zero angle of attack and confirm that air speed is correct.
   You may have to adjust fan speed to account for blockage created by the wing.
6. Record pressure readings for all the pressure tappings
7. Plot a line of pressure vs chord position on the provided chart.
8. Find and record the angle of attack α_s at which stall occurs, and repeat steps 5 to 7.
   A good indicator are the tufts attached to the wing suction side.
9. Select reasonable number of pre-stall angles of attack ($<\alpha_s$) and post-stall angles of attack ($>\alpha_s$), and repeat steps 5 to 7. The selected angles of attack should enable you to plot and show the changes/trends in lift and drag coefficients for different angles of attack.
10. Select reasonably higher speed to account for the effects of high speeds (please note the fan speed should not exceed 1000 rpm), then repeat steps 4. to 9. During the process, check repeatability by returning to at least one setting and taking a second set of readings.
11. *(Optional)* Determine the sensitivity of α_s to U. (Choose values of speed, U that you think are appropriate. Feel free to experiment carefully but do not exceed the maximum pressure difference of 500 Pa).

Analysis Tasks

1. From measurements of pressure difference, ambient static pressure and ambient temperature calculate U.

2. Directly plot the measured pressures vs chord length on a chart similar to the one shown in Fig. 8. Pre-printed copies will be available from the tutors / on Blackboard

3. Convert pressures to pressure coefficient C_p and re-plot selected pressure distributions on a single graph. (using Excel or Python helps)

4. Integrate the pressure distributions to obtain the wing normal force, N and then calculate lift and drag.
   It is possible to integrate the pressure numerically by considering the area within the pressure vs chord length curve. This can be done most simply using the trapezoidal rule from MECH2700.

5. Repeat steps 2 - 4 for selected combinations of speed and angle of attack.

6. Discuss the relationship between the C_p curves for the different speeds and different angles of attack.

7. Discuss the relationship between the C_L and C_D curves for the different speeds.

Report

As a group, write an brief engineering report summarising the laboratory experiment and any findings. The report should include the following:

- A brief introduction on the goals of the experiment.
- Description and analysis of the measurement uncertainties. This analysis should give an insight on the quality of the measured data. Similarly, during discussion of results you should consider how measurement errors can influence your results.
- The Results and Analysis section should include data specified in the Experimental Procedure and the results of Analysis Tasks 1-5 above.
- As part of the Discussion, respond to Analysis Tasks 6 and 7 above and address the following questions:
  - How does boundary layer growth rates vary between top and bottom of the aerofoil.
  - Is there a pressure difference across the boundary layer? Why/Why not.
  - Is the flow on the wing surface expected to be laminar or turbulent? What happens to the flow downstream of the separation?
  - What are appropriate numerical methods to integrate the pressures acting on the wing?
  - What pressure can be assumed for the wing trailing edge?
  - Where is the stagnation point. What do the pressure measurements tell you about the stagnation point location?

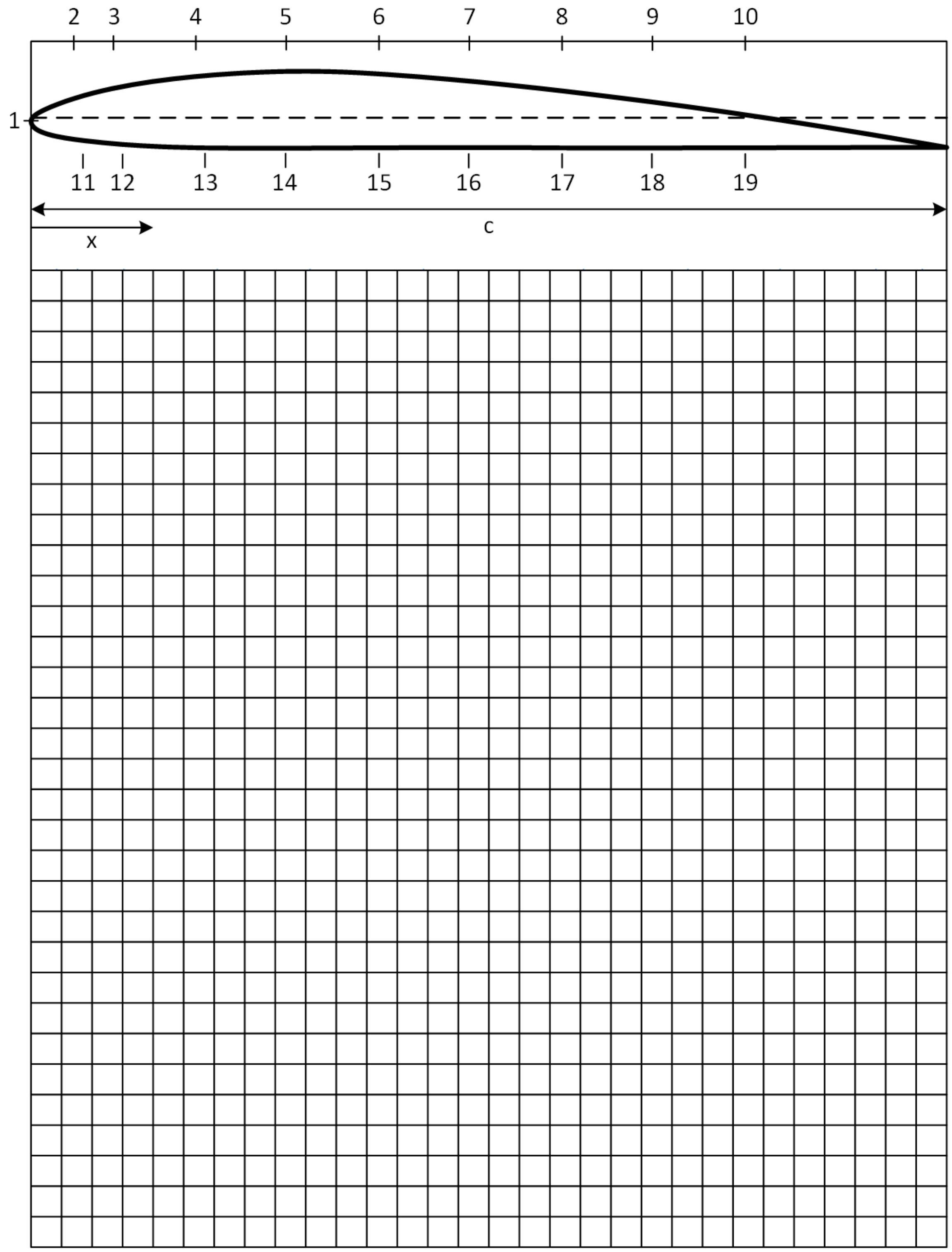

Figure 8: Wing Profile with Pressure Tapping Locations