Document in Progress

AAE 33401 Lab Manual
Aerodynamics Lab

Draft: Prepared in Fall 2023 - Spring 2024

Last Updated: August 13, 2025

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Contents

Chapter 1

Airfoil Characteristics

1.1 Introduction

Airfoils are specialized shapes used in wings, blades, and other surfaces to generate lift as air flows over them. The design of an airfoil is crucial in determining the efficiency and performance of an aircraft or turbine. High-lift devices, such as flaps and slats, are employed to enhance the lift generated by an airfoil, especially during critical phases of flight like takeoff and landing. These devices alter the airfoil's shape and airflow characteristics, allowing for greater lift at lower speeds, improving safety and performance in various flight conditions.

1.2 Lab Overview

In this lab, we will investigate the lift and drag characteristics of a cambered airfoil (NACA 2415) with and without high-lift devices (trailing edge flap and leading edge slat). First, we will measure the wind tunnel speed vs. the motor frequency using a pitot probe. Then, the airfoil models will be mounted to a force balance in the low-speed wind tunnel. Load cells in the force balance allow us to measure the lift and drag on the airfoil as we increase the airfoil angle of attack. The force data is read via LabVIEW software interface. The lift coefficient vs. angle of attack and drag polar (drag coefficient vs. lift coefficient) will be compared for (1) the base airfoil, (2) the airfoil with a trailing edge flap deployed, and (3) the airfoil with both a trailing edge flap and leading edge slat deployed.

1.3 Objectives

The purpose of this experiment is to study the aerodynamic characteristics of typical low subsonic airfoil profiles. Several different airfoil profiles are available for study. During this experiment, you will:

1. Measure lift and drag versus angle of attack for various airfoil sections.
2. Measure the effect of flaps and leading edge slats.
3. Find the stall angles for the airfoils.

1.4 Background

After a century of theoretical research on the subject of airfoil and wing theory, the final word on the performance of an airfoil must still come from wind tunnel testing. The reason for this state of affairs is that the flow field about a wing is extremely complicated. The simplifying assumptions that are frequently introduced in order to treat the problem theoretically are much too severe to fail to influence the final results. Many of these assumptions ignore the effects of viscosity, non-linearities in the equations of motion, three-dimensional effects, unsteady flow, free stream turbulence, and wing surface roughness. Nevertheless, the theoretical prediction of lift produced by a wing has been reasonably successful (not quite so true for drag) and serves as an effective basis with which to study the experimental results.

1.4.1 Aerodynamic Coefficients

Results of two-dimensional airfoil theory and experiment are presented in terms the lift coefficient. Their definitions are summarized in equations 1.1, 1.2 and 1.3.

$$C_l = \frac{L/unitspan}{\frac{1}{2}\rho_\infty U_\infty^2 c} \tag{1.1}$$

$$C_d = \frac{D/unitspan}{\frac{1}{2}\rho_\infty U_\infty^2 c} \tag{1.2}$$

$$C_m = \frac{M/unitspan}{\frac{1}{2}\rho_\infty U_\infty^2 c^2} \tag{1.3}$$

The variables in the above equation carry the following meanings:
C_l = Lift Coefficient
C_d = Drag Coefficient
C_m = Pitching Coefficient
L = Lift
D = Drag
M = Pitching moment
ρ_∞ = Freestream density
U_∞ = Freestream Velocity
c = Chord length

Figure 1.1 shows the various forces and important locations on an airfoil. Note that the lift force (L') acts perpendicular to the direction of free stream velocity and drag force (D') acts in the direction of free stream velocity (V_∞).

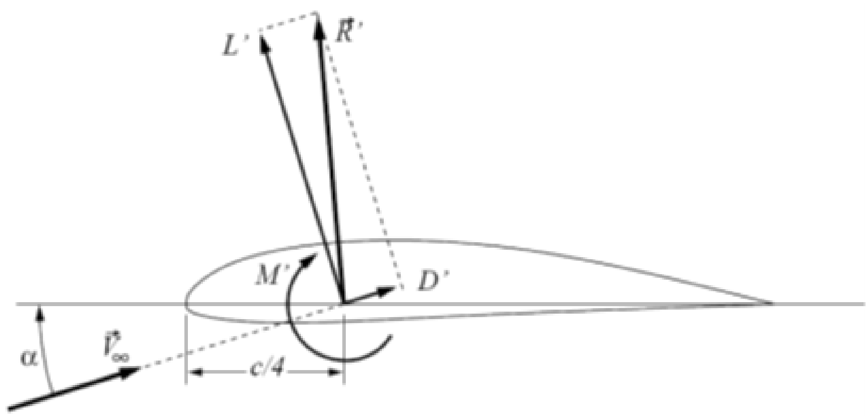


Figure 1.1: Airfoil labelled with important location and vectors. Note: It is common to place the chord line on the x axis and let the free stream direction vary with angle of attack. This helps with geometry definition in CFD analysis.

Figure 1.2 shows a sketch of a typical airfoil and the reasons for the round leading edge (to prevent leading edge separation over a range of angles of attack) and the sharp trailing edge (to set the rear stagnation point and reduce base drag).

1.4.2 Airfoil Geometry

The geometry of an airfoil is defined in terms of the thickness, chord, and the detailed coordinates of the upper and lower surfacess. The camber line is half-way between the upper and lower surfaces. Figure 1.3. shows the airfoil geometry definitions.

1.4.3 Thin Airfoil Theory

Thin airfoil theory replaces the airfoil with a distribution of vortices (Figure 1.4) on the camber line. (Anderson Chapter 4)

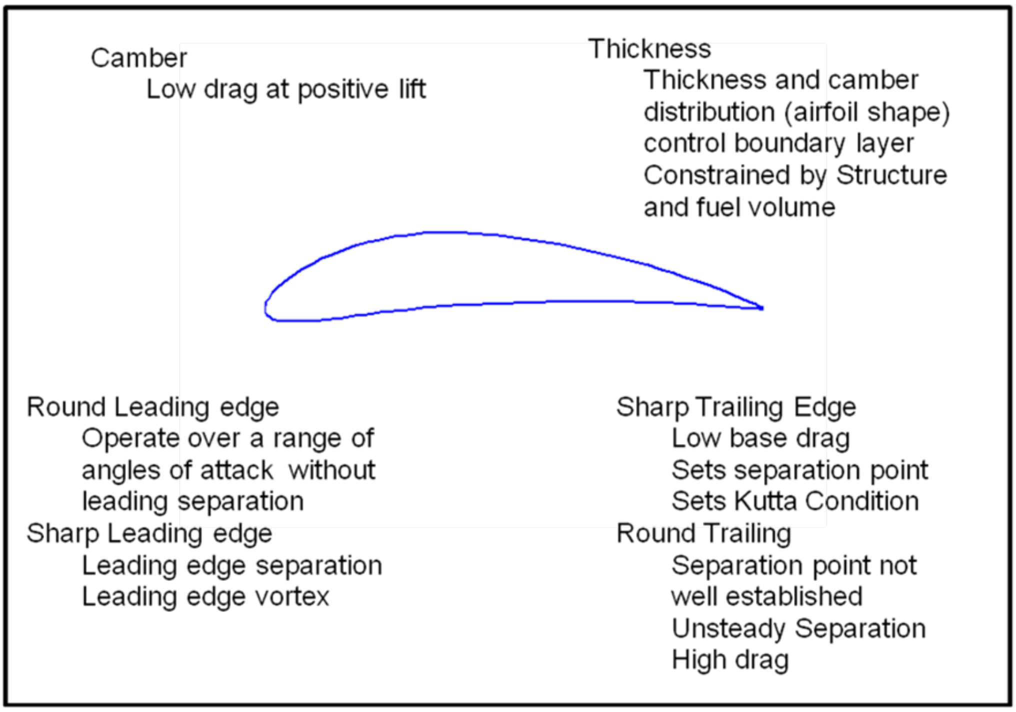


Figure 1.2: Anatomy of an Airfoil

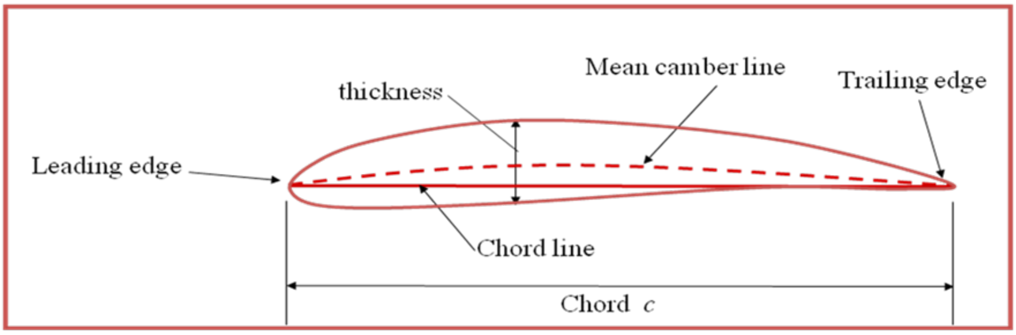


Figure 1.3: Airfoil Geometry

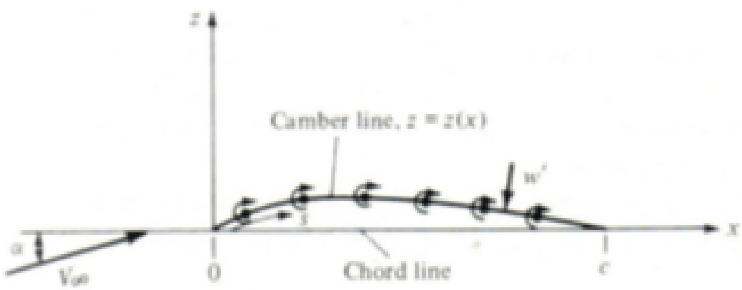


Figure 1.4: Vortex Sheet on Camber Line

This simple theory gives the important results below:

- $C_l = 2\pi(\alpha - \alpha_0)$
- $\frac{\partial C_l}{\partial \alpha} = 2\pi$: This is the slope of the lift curve
- $C_d = 0$: d'Alembert's Paradox
- $\frac{\partial \mathrm{C_m}}{\partial \alpha} = 0$ for symmetric sections. Center of Pressure at $x = c/4$
- $\frac{\partial \mathrm{C_m}}{\partial \alpha} =$ negative constant for cambered sections. Center of Pressure at $x > c/4$

Actual results deviate from thin airfoil theory at high angles of attack as the airfoil stalls.

1.4.4 NACA Experimental Results

NACA Nomenclature For Airfoils

Most of the NACA airfoils are classified among three types: the four-digit, the five-digit, and the series 6 sections. The meanings of these designations are illustrated by the examples below.

- **NACA 4415**
  - 4 - The maximum camber of the mean line is 0.04c. (first digit)
  - 4 - The position of the maximum camber is at 0.4c. (second digit)
  - 15 - The maximum thickness is 0.15c. (third and fourth digits)
- **NACA 23012**
  - 2 - The maximum camber of the mean line is approximately 0.02c. The design lift coefficient is 0.15 times the first digit for this series. (first digit)
  - 30 - The position of the maximum camber is at $0.30/2 = 0.15$c. (second and third digits)
  - 12 - The maximum thickness is 0.12c. (fourth and fifth digits)
- **NACA $65_3 - 421$**
  - 6 - Series designation. (first digit)
  - 5 - The minimum pressure is at 0.5c. (second digit)
  - 3 - The drag coefficient is near its minimum value over a range of lift coefficients of 0.3 above and below the design lift coefficient (subscripted number).
  - 4 - The design lift coefficient is 0.4. (first digit after hyphen)
  - 21 - The maximum thickness is 0.21c. (last two digits)

For further information, see *Theory of Wing Sections*, by Ira H. Abbott & Albert E. Von Doenhoff, Dover Publications, Inc., New York, 1959. Experimental results from Abbott and Von Doenhoff are shown in figure 1.5. Figure 1.5 is data for the NACA 2415 airfoil which we will test in this lab. Note that increasing Reynolds number delays the stall and reduces drag slightly. Also, note that the surface roughness causes a significant increase in drag.

Figure 1.6 shows data for the laminar flow airfoil NACA $65_2 - 415$. This airfoil has laminar flow on the front portion of the airfoil boundary layers for lift coefficients from 0 to 0.8. The drag is reduced by almost a factor of two in this "drag bucket" region. Note the shift in the lift curve to the left when the split flap is deployed.

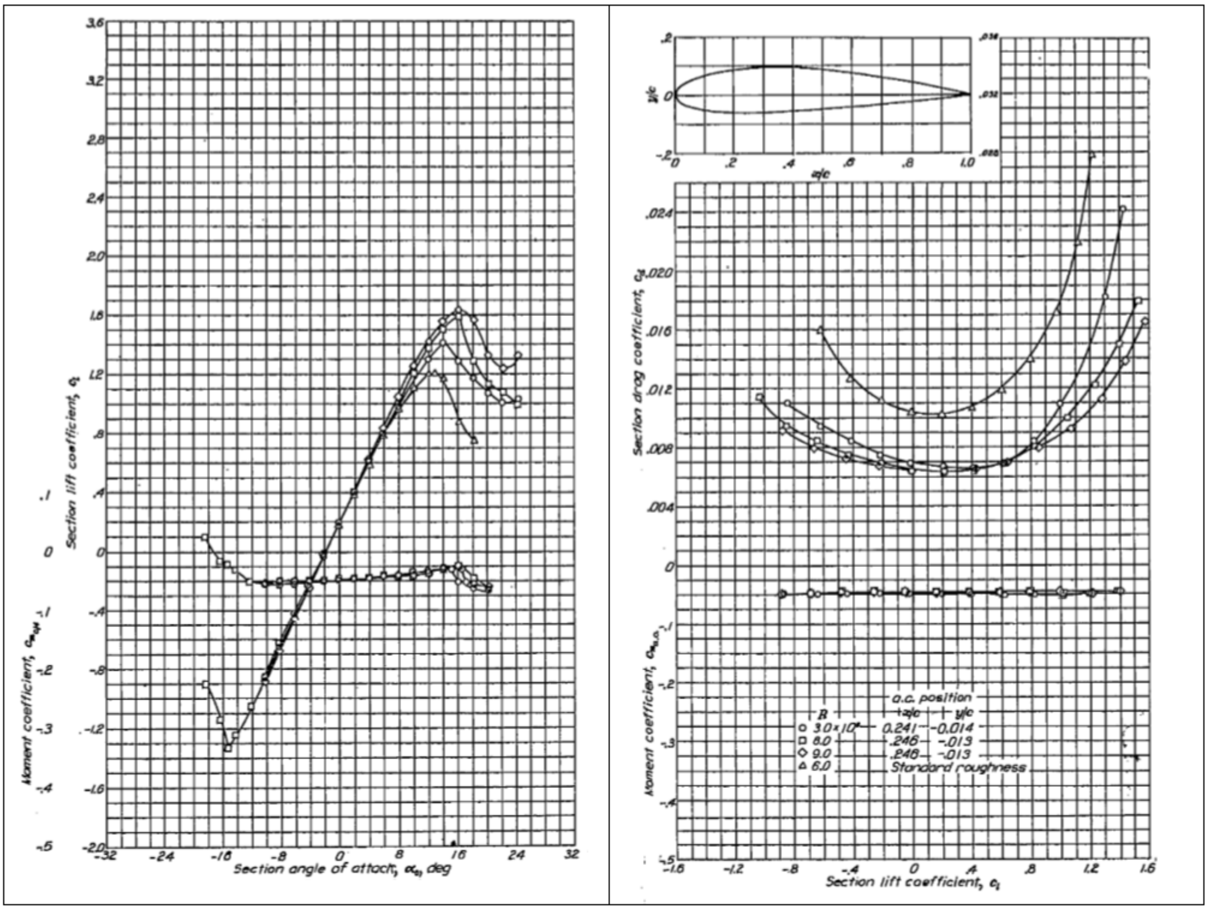


Figure 1.5: NACA Experimental Results for the NACA 2415 Airfoil|Enlarged Image: Sec: 1.8.2

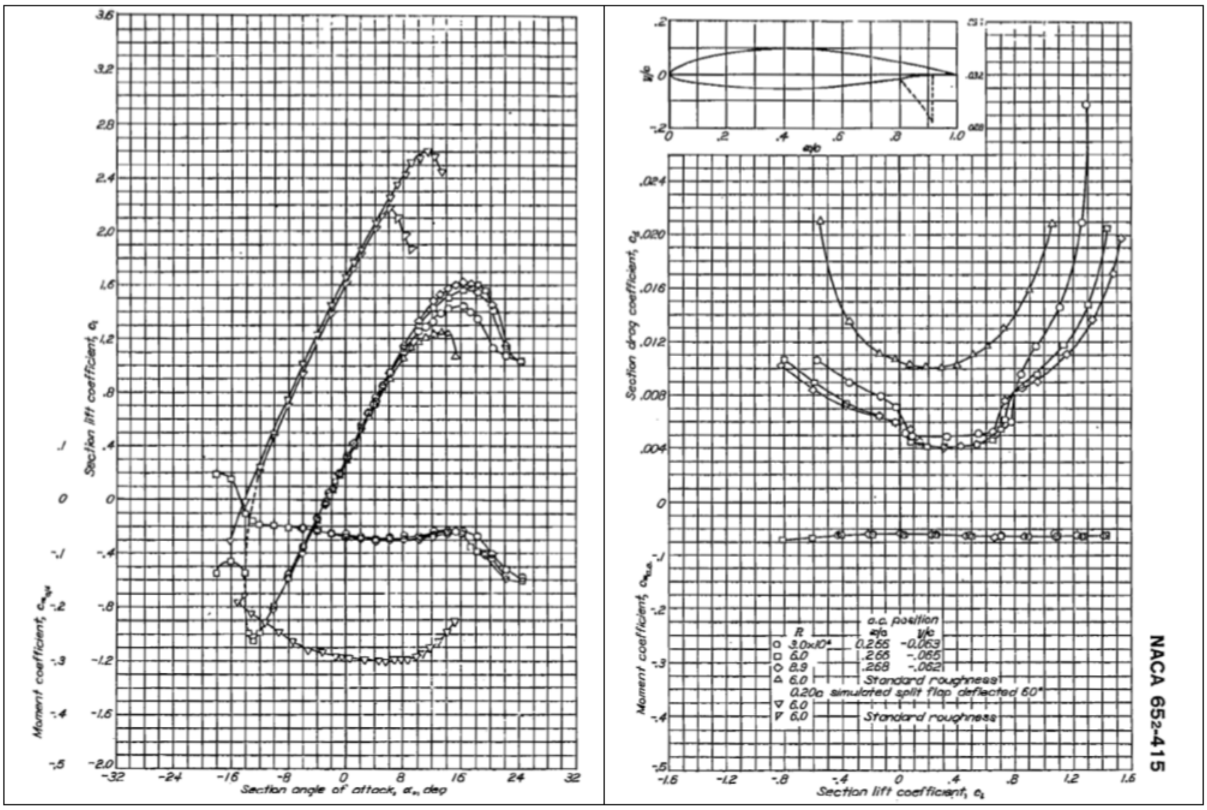
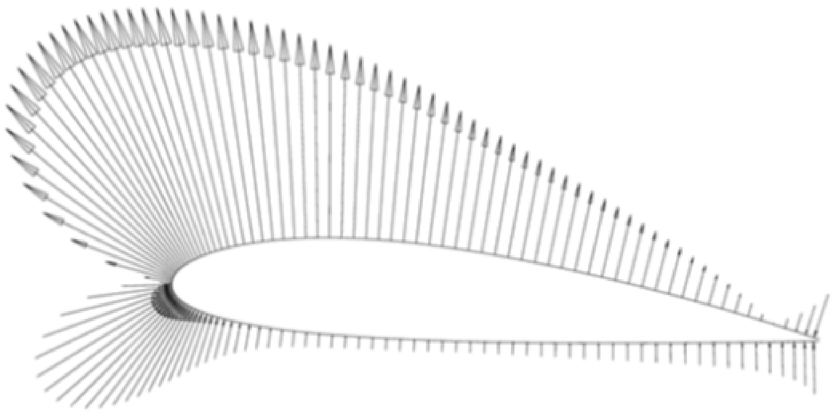


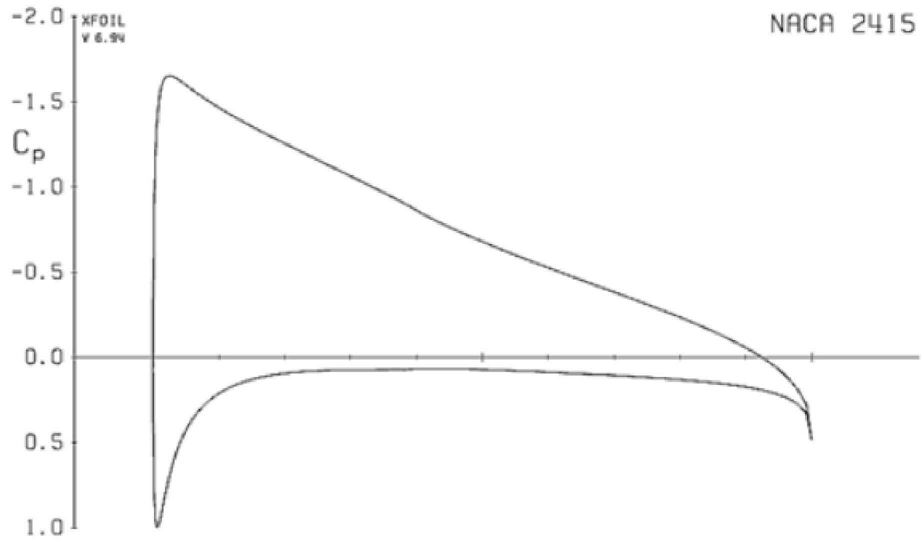
Figure 1.6: NACA Experimental Results for the NACA $65_2 - 415$ Airfoil

1.4.5 Pressure Distribution

The pressure distribution on the NACA 2415 airfoil at 5 degrees angle of attack as calculated by X-Foil is shown in Figure 1.7.



(a) Force Vectors due to Pressure Acting on airfoil Surface



(b) Pressure Coefficient Distribution

Figure 1.7: Pressure Distribution on a NACA 2415 Airfoil at 5 degrees angle of attack

1.4.6 Flaps and Slats

When you need to go slow for takeoff and landing, a high lift coefficient is needed so lift can still balance weight. The higher the maximum C_L, the slower you can fly. Flaps increase camber, shift the lift curve to the left and in some configurations increase the wing area. (Figure 1.8). Slats, one type of leading edge device, help avoid stall near the leading edge and increase the maximum lift coefficient.

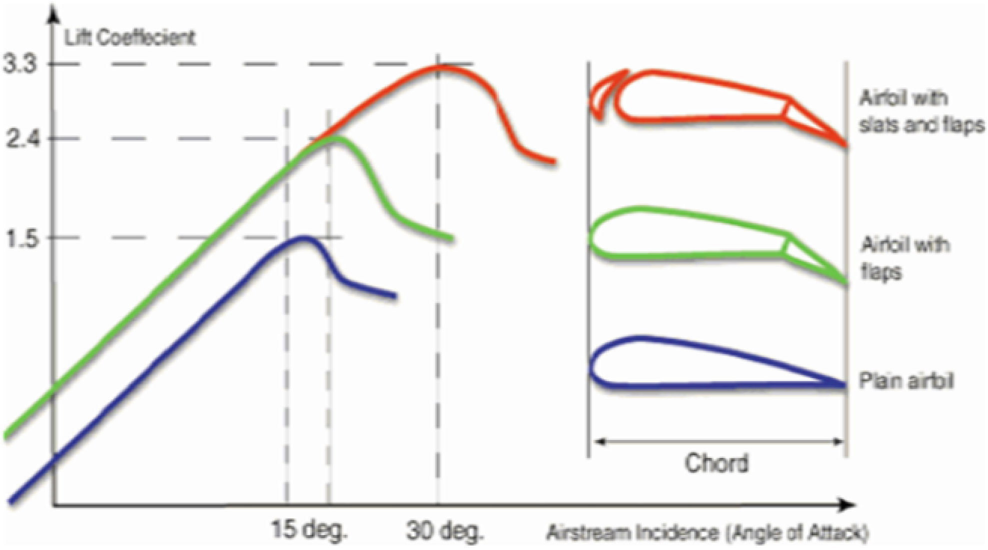


Figure 1.8: Effect of Flaps and Slats on Lift Coefficient

1.5 Recommended Reading

It is highly recommended that students read the following sections in Fundamentals of Aerodynamics by John D. Anderson:

- 4.2 Airfoil Nomenclature
- 4.3 Airfoil Characeristics
- 4.11 Modern Low-Speed Airfoils
- 4.13 Applied Aerodynamics: The Flow Over an Airfoil - The Real Case

1.6 Data to be Acquired

Following is the data that you need to acquire over the course of this lab:

1. Actual mass of the physical weights used
2. Mass of the physical weights from LabVIEW
3. Dynamic pressure inside the wind tunnel during operation
4. Micrometer value for change in angle of attack
5. Dimensions of airfoils used (for obtaining the coefficients)
6. Lift and Drag Readings for various angles of attack

1.7 Lab Setup and Procedure

1.7.1 Experimental Setup and Preparation

Lab Location: ARMS B098, Low Subsonic Wind Tunnel
In this experiment, lift and drag on an airfoil and the effect of flaps and slats will be studied using the Armstrong subsonic wind tunnel (ARMS B098). Measurements from force transducers will be recorded using LabVIEW software to calculate the actual lift and drag forces over a range of angles of attack. A schematic of the set-up in the Armstrong wind tunnel is shown in Figure 1.9.

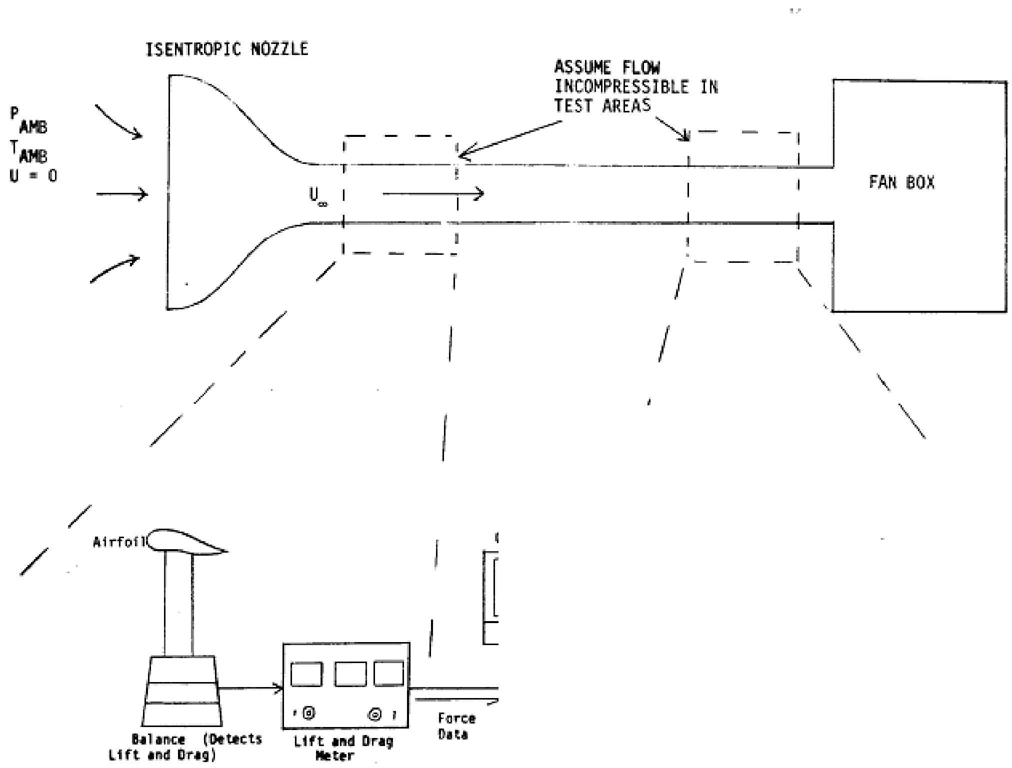


Figure 1.9: Schematic of Armstrong subsonic wind tunnel with airfoil stand

Description of Airfoils

A typical airfoil used in the Airfoil Characteristics Experiment is shown in Figure 1.10. It has a finite length, and detachable endplates. With the endplates attached, it is assumed that the airfoil behaves like a two-dimensional wing (*can you justify this assumption?*). A single element airfoil (NACA 2415), two element airfoil (main element plus flap) and a three element airfoil (main plus slat plus flap) will be tested.

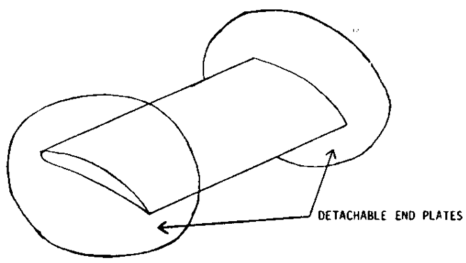


Figure 1.10: Airfoil with End Plates

Preparation

1. ☐ One member should log into the lab computer using their career account and navigate to the Lab 1 folder at the destination: `C:\ Temp \ AAE33401 \ Lab 1 - Airfoil Characteristics`. Make sure you save any and all data generated during the lab in the `data_33401_lab1` folder. Make a folder with your team name (e.g. Black16) within that folder.

2. Download the Lab 1 data spreadsheet from Brightspace, and open it on the computer. The spreadsheet will help you record data and perform data analysis.

3. Open the LabVIEW VI (virtual instrument)"**Lab1_33401_WindTunnelBalance**". Click here to learn more about LabVIEW. **Please do not alter or modify any LabVIEW VIs - even very small changes can cause the program to stop working!**

4. Set the file path in the LabVIEW VI to your data folder. After the path, add a backslash and enter a name for the data file (in .txt format). For example, your final path might look like: `C:\temp\AAE33401\Lab 1 - Airfoil Characteristics\data_33401_lab1\Black16\Black16_airfoil_data.txt`.

5. Click the Run arrow at the top toolbar to begin running the program. The readouts from the force balance should be fluctuating near zero. If the readouts are not fluctuating or there is an error message, get help from the TA.

6. Verify that the program saves the data correctly as follows:

   6.1 Click the "Push to Write" button in the VI to write the force to the file you specified in the LabVIEW path. A push is confirmed when you see the green "Success" light and when the push counter increases.

   6.2 Press the STOP button on the VI to stop the program (NOT the stop sign on the top toolbar but the word STOP above the file path).

   6.3 Open the data file using Notepad, Wordpad or Excel. The first row of data is the column heading ("Micrometer", "Lift", "Drag", "F0", "F1", "F2", "F3"). Note that F0-F3 are the raw force readings directly from the load cells in the force balance (see Section section 7.1) but the VI already uses these values to calculate the lift and drag (in lbf) for you. The second row in the data file is the readings at the time you clicked the "Push to Write" button.

1.7.2 Experimental Procedure

Yellow bold highlighted text indicates that student should show the data/plot obtained to the TA present at the lab.

1. **Calibration of Force Balance**

   1.1 ☐ **Calibrate the Lift**

      1.1.1 ☐ Make sure the tunnel is not running, contains no test specimen in the test section and that there are no weights placed on the force balance.

      1.1.2 ☐ Run the WindTunnelBalance VI. When you run the program, it zeroes out all the forces in the sensors.

      1.1.3 ☐ Record the initial values of lift in data spreadsheet, against zero actual weight (when no weights are placed on the balance) from the VI.

      1.1.4 ☐ Apply known (total) weights of 0.5, 1, 2, 3, 4 and 5 lbs to the middle of the top plate of the force balance. Start with the 0.5 lb weight and then stack additional weights on top. Wait for the reading from the force balance to even out, then record the data from LabVIEW in the Lab 1 data spreadsheet. Note that we are applying a downward force on the balance and so we will read a negative value of lift from LabVIEW. For simplicity, record both the applied and measured weights as positive values.

      1.1.5 ☐ Plot the "Actual Weight" vs "Measured Weight" for lift in the data spreadsheet. Note that the curve is quite linear for the most part, but remember that calibration is imperative to get accurate data.

      1.1.6 ☐ Fit a linear calibration curve to the lift data. Record the values of the slope and intercept of the calibration curve in the appropriate cells in the worksheet. **Show your calibration curve to the TA before proceeding.**

   1.2 ☐ **Calibrate the Drag**

      1.2.1 ☐ Remove the weight(s) used to calibrate the lift.

      1.2.2 ☐ Repeat steps 1.1.1 to 1.1.5 to calibrate the drag, but instead of placing weights on the top balance plate, place them on the pulley platform located just behind the F2 transducer.

      1.2.3 ☐ Plot the "Actual Weight" vs "Measured Weight" for drag in the data spreadsheet.

      1.2.4 ☐ Fit a linear calibration curve to the drag data. Record the values of the slope and intercept of the calibration curve in the appropriate cells in the worksheet. **Show your calibration curve to the TA before proceeding.**

2. ☐ Mount the airfoil stand inside onto the force balance. Ensure that the stand is installed tightly and that all openings in the tunnel test section are sealed.

3. ☐ Stop the VI and run it again, to make sure all forces are zeroed out. ***Make sure you always start the VI before turning on the tunnel to start with zero forces when there is no flow.***

4. ☐ Turn on the wind tunnel following the Wind Tunnel Operation Procedure given in section 7.1. Set the wind tunnel speed to 30 Hz.

5. ☐ Turn on the pressure differential meter (digital manometer) and note the pressure reading. Use the differential pressure to calculate the tunnel free-stream velocity in the data spreadsheet. **Show your result for the free-stream velocity to the TA.** Free stream velocity will be needed for calculating the non-dimensional factor, which is used when obtaining the coefficient of lift.

6. ☐ Take lift and drag readings without any airfoil on the stand (to get a starting reading of the lift and drag on just the stand). Note them in the data spreadsheet.

7. ☐ Turn off the tunnel, remove the tunnel lid and insert the **airfoil with flaps**. Measure the span and chord of the airfoil using the metal ruler (check if the values in the data spreadsheet are correct). Tightly secure the airfoil to the stand using the screw mounts. Use a level to set the angle of attack of the

airfoil to zero. Turn on the micrometer by pressing the ON button and zero out the micrometer value by long-pressing the power button.

8. ☐ Replace the tunnel lid and turn on the tunnel. Remember to start the LabVIEW program before starting the tunnel for each set of runs as this zeroes the forces.

9. ☐ Take lift and drag measurements for angles of attack from 0 to 20 degrees in approximately two degree increments by adjusting the micrometer on the stand.

   9.1 ☐ Calculate the micrometer value required for obtaining a 2 degree increment in angle of attack. Use this increment to calculate the micrometer height vs. angle of attack in millimeters and record in the table in the data spreadsheet. You can refer to this table for changing the angle of attack during measurements. Learn more about correlating the angle of attack with micrometer measurements in section 7.1.4.

   9.2 ☐ Enter the micrometer reading in LabVIEW (to keep track of the angle of attack for a given measurement).

   9.3 ☐ Press "Push to Write" to record the data for lift and drag for the set angle of attack in your text file.

   9.4 ☐ Change the angle of attack by adjusting the micrometer and update the micrometer reading in LabVIEW. Click "Push to Write" to record the lift and drag.

   9.5 ☐ Repeat the previous step up to 20 degrees angle of attack.

10. ☐ Calculate the lift and drag per unit span by dividing the force by the span of the airfoil you measured earlier. Remember to convert the lift and drag units from pounds force to Newtons and the span measurement to meters.

11. ☐ Calculate the lift and drag coefficients c_l and c_d by dividing the lift and drag per unit span by the dynamic pressure times the airfoil chord length ($\frac{1}{2}\rho_\infty V_\infty^2 c$). Fill this value in the "non-dimensionalizing factor" field. Make sure your units are consistent!

    Show your plots the lift and drag coefficients vs. angle of attack for the first airfoil to the TA before proceeding.

12. ☐ Turn off the tunnel, remove the lid and the airfoil and repeat with the other two airfoils - **one with a flap only and one with both a flap and a slat**. For the airfoil with the flap, fix the flap at the maximum deflection (approximately 35°). We suggest adjusting the micrometer back to zero without an airfoil mounted (it goes quicker) and use the level to quickly confirm that the angle of attack is indeed zero before proceeding. **Chord and span of the other two airfoils are the same as the base airfoil**.

13. ☐ **Return the experiment and the lab to its state when you first arrived.** Return all tools and equipment back to their designated location, turn off any equipment that was started for the purpose of your lab, and clean up your workspace (you may leave the PC on, and sign out of your account instead). **Obtain approval from the TA before leaving the lab.** Thank you!

***** ALWAYS TURN OFF THE TUNNEL BEFORE OPENING LID TO CHANGE OBJECT IN TUNNEL.***
***** ALWAYS CLICK "RUN" IN THE LABVIEW VI BEFORE TURNING ON THE TUNNEL.***

1.8 Additional Information

1.8.1 Airfoil Design

There are multiple airfoil with different configurations that are used in this lab. Figures 1.11 and 1.12 are the schematics of the airfoil used in the lab.

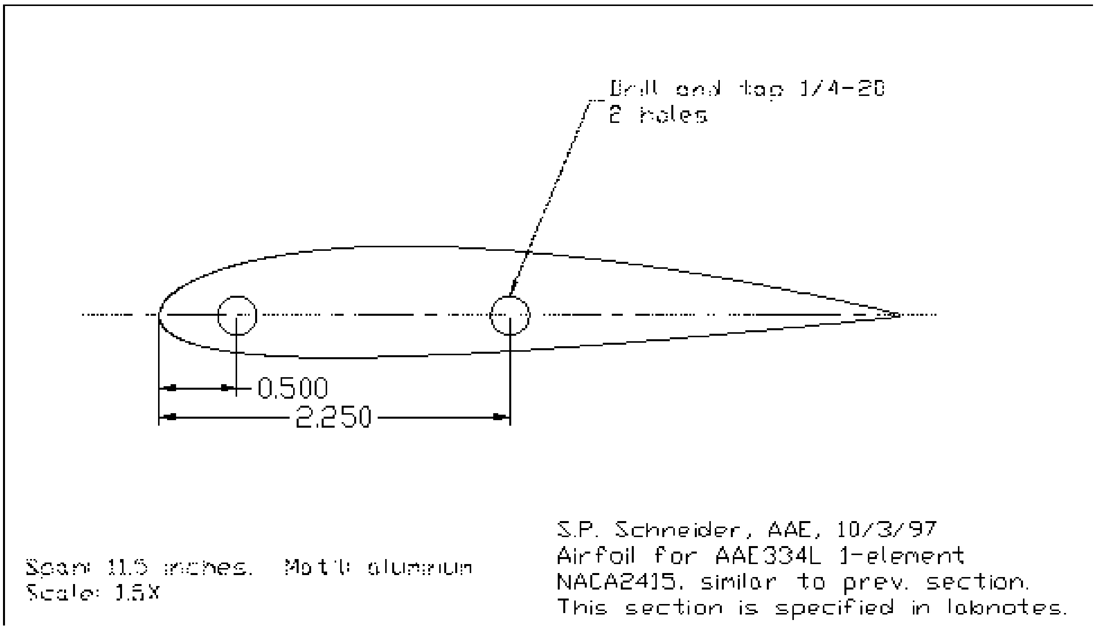


Figure 1.11: Drawing of a single element airfoil

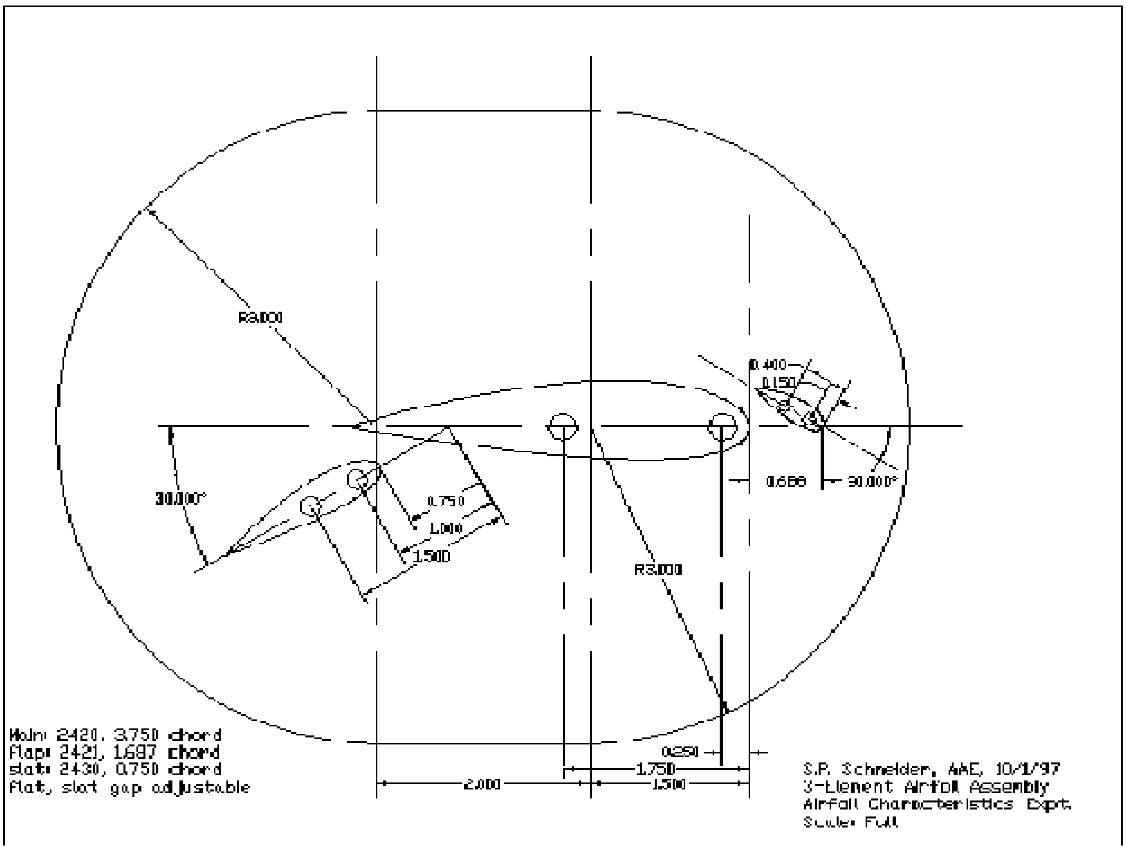
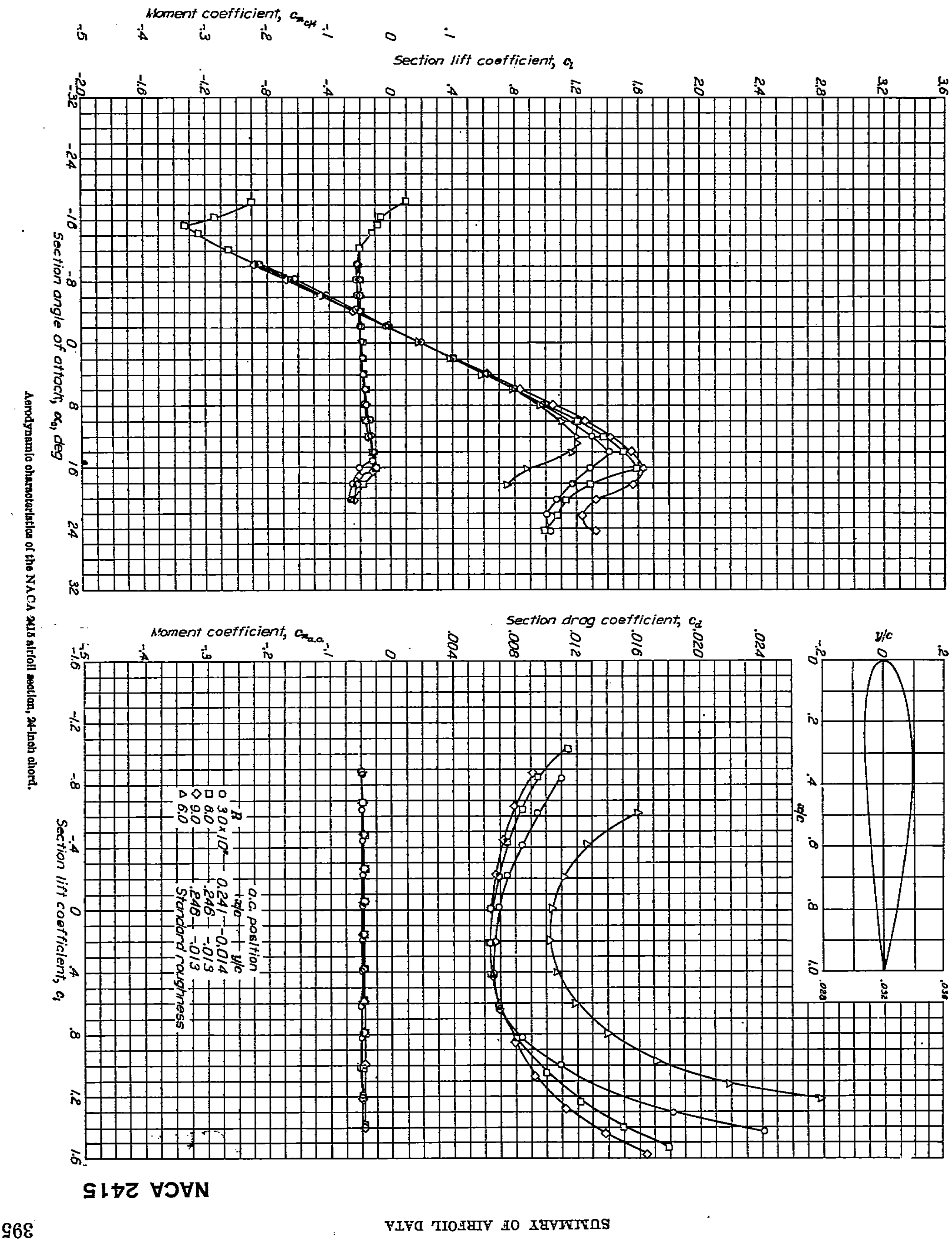


Figure 1.12: Airfoil with flaps and slats

1.8.2 NACA 2415 Plots of Experimental Results



Chapter 2

Pressure Tap Airfoil

2.1 Introduction

Surface pressure measurements on airfoils and wings are vital in understanding and optimizing the aerodynamic performance of aircraft. These measurements provide detailed insights into how pressure is distributed across the surface of an airfoil or wing as air flows over it. This pressure distribution directly influences the lift, drag, and overall stability of an aircraft, making it a critical factor in aerodynamic design. By analyzing surface pressure data, engineers can identify regions of high and low pressure, enabling them to refine airfoil shapes, improve efficiency, and reduce adverse effects like flow separation or turbulence. These measurements are particularly important in the design and testing of high-performance aircraft, where even minor deviations in pressure distribution can lead to significant changes in flight characteristics. Surface pressure data also play a crucial role in validating computational fluid dynamics (CFD) models, ensuring that simulations accurately reflect real-world conditions. Furthermore, in the development of high-lift devices, surface pressure measurements help in assessing the effectiveness of flaps, slats, and other mechanisms in enhancing lift during takeoff and landing. Overall, surface pressure measurements are an indispensable tool in the continuous quest for safer, more efficient, and more advanced aeronautical designs.

The flow characteristics over an airfoil are based on the change of angle of attack and free stream velocity. In order to analyse the change in pressures, pressure taps are installed on the airfoil body. By reading pressure values, coefficients of pressure, lift and drag can be calculated for plotting appropriate curves.

2.2 Lab Overview

In Lab 1 we measured the total forces on an airfoil using a force balance and calculated the lift and drag coefficients vs. angle of attack. In this lab, we will measure the pressure distribution on a NACA laminar flow airfoil surface using an airfoil model with surface taps. The pressure at each tap will be recorded simultaneously using LabVIEW. The surface pressure distribution will then be used to estimate the lift and drag coefficient vs. angle of attack and compared with historical data.

2.3 Objectives

In this experiment, you will:

1. Measure the pressure distribution over the airfoil.
2. Observe the transition point from laminar to turbulent flow, and how it changes with angle of attack.
3. Find the stall angles for the airfoil.
4. Find the lift curve for the airfoil.

2.4 Background

The purpose of this experiment is to study the pressure distribution over a laminar flow airfoil, observe how the flow changes with angle of attack, and compare the data to theoretical data in XFOIL.

2.4.1 Discussion

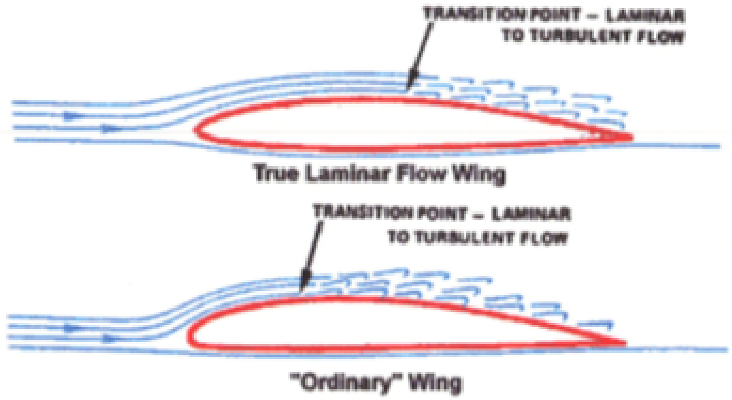


Figure 2.1: Comparison of transition for a laminar vs. ordinary airfoil. *Courtesy: allstar.fiu.edu*

Flow around bodies is an incredibly complex system. Even with advancements in computational fluid dynamics, wind tunnel testing is still needed to fully evaluate airfoil performance. Laminar flow is a smooth, uninterrupted flow of a fluid. As the fluid transitions to turbulent flow, it has a more erratic and chaotic flow field. A schematic of transition on an airfoil is shown in Figure 2.1. Laminar flow airfoils were created to prolong the transition from laminar to turbulent flow. The turbulent flow can result in a reduction of lift and also an increase in drag due to increased skin friction on the airfoil. The NACA airfoil used for this lab is a NACA $66_2 - 215$, shown in Figure 2.2, and is a laminar flow airfoil.

2.4.2 Airfoil Geometry

The geometry of an airfoil is defined in terms of the thickness, chord, and the detailed coordinates of the upper and lower surface. The camber line is half way between the upper and lower surface. Figure 2.3 highlights the important airfoil geometry definitions.

Laminar flow airfoils are quite different in appearance from a "normal" airfoil. The airfoil is usually thinner than a conventional airfoil, the leading edge is often more pointed, and the airfoil is typically closer to symmetric. Another main difference is that the maximum thickness for a laminar flow airfoil typically occurs near the 0.5c position as opposed to the 0.25c position of a conventional airfoil. These differences can be seen by comparing the airfoils in Figures 2.2 and 2.3.

2.4.3 NACA $66_2 - 215$aminar Flow Airfoil

The airfoil used in the experiment is a 6-series NACA airfoil. The 6-series airfoils were designed mathematically to have a specific location along their chord where the flow will transition from laminar to turbulent flow. Laminar flow airfoils can provide advantages such as low drag over a small operating range and a high (C_L)max.

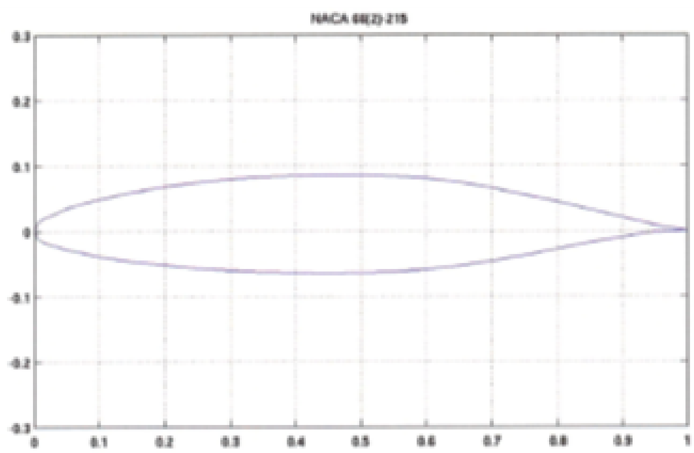


Figure 2.2: NACA $66_2 - 215$ airfoil taken from the UIUC Airfoil Database.

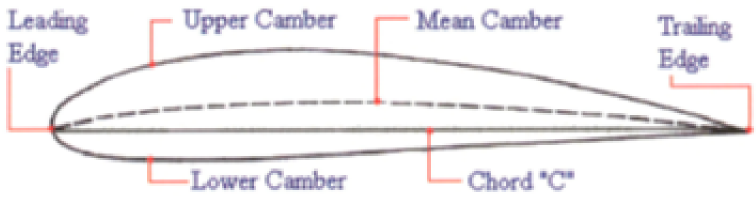


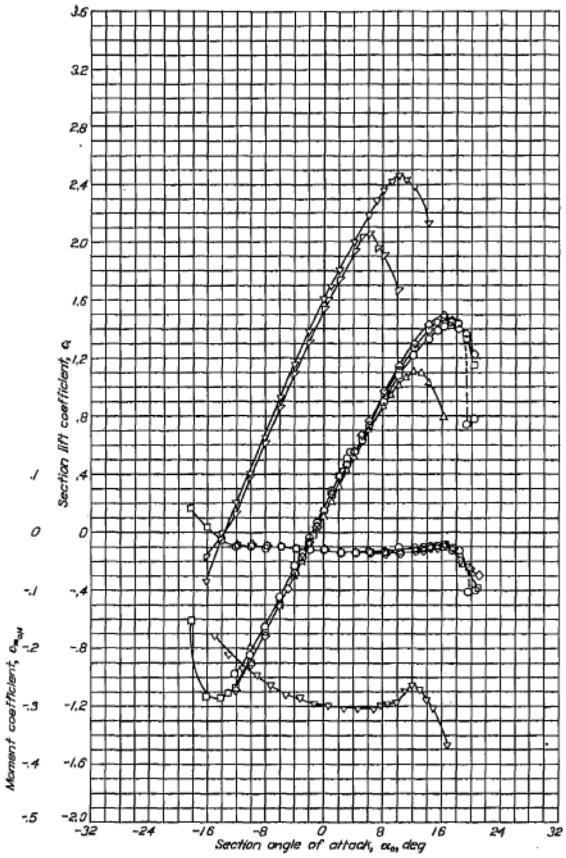
Figure 2.3: Airfoil Geometry. *Courtesy: Airplane Aerodynamics by Dommasch, Sherby, and Conally, Pitman Press, 1967.*

The disadvantages of these airfoils are that they have poor stall characteristics, high pitching moment, and a very thin trailing edge. The nomenclature for a six-digit airfoil is given below. **NACA** $66_2 - 215$

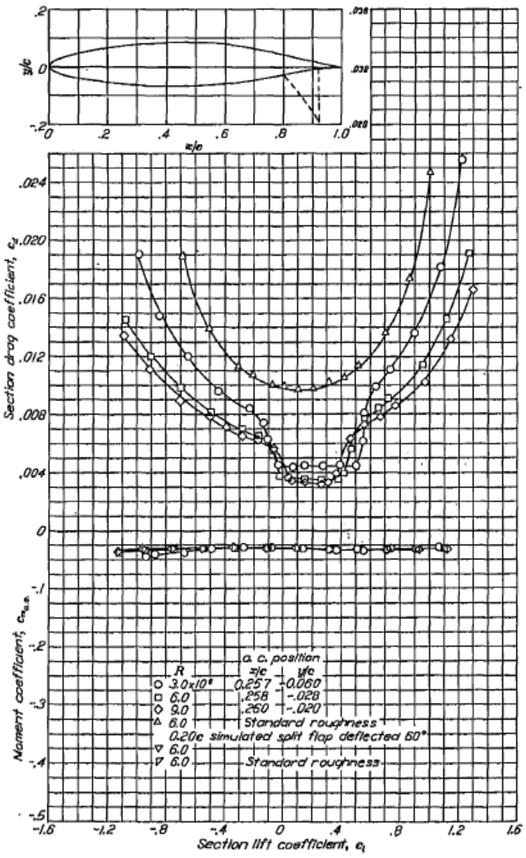
- 6 - Series designation. (first digit)
- 6 - The minimum pressure is at 0.6c. (second digit)
- 2 - The drag coefficient is near its minimum value over a range of lift coefficients of 0.2 above and below the design lift coefficient (subscripted number).
- 2 - The design lift coefficient is 0.2. (first digit after hyphen)
- 15 - The maximum thickness is 0.15c. (last two digits)

Figure 2.4a and 2.4b shows historical wind tunnel test data for the laminar flow airfoil NACA 66_2-215[1]. . This airfoil has laminar flow on the front portion of the airfoil boundary layers for lift coefficients from 0 to 0.8. The drag is reduced by almost a factor of two in this "drag bucket" region. Note the shift in the lift curve to the left when the split flap is deployed.

[1]Airfoil data from "Theory of Wing Sections" by I. H. Abbott and A. E. von Deonhoff (1959), pp. 668-669



(a) Lift coefficient and moment coefficient (about the quarter-chord) vs. angle of attack.



(b) Drag polar (drag coefficient vs. lift coefficient) and moment coefficient (about the aerodynamic center).

Figure 2.4: Wind tunnel data for the NACA $66_2 - 215$ laminar flow airfoil ((from Abbott and von Doenhoff[1]).

2.4.4 Pressure Distribution

The pressure distribution on the NACA $66_2 - 215$ airfoil at 5 degrees angle of attack can be calculated using XFOIL. The resulting pressure vectors and pressure coefficient are shown in Figure 2.5.

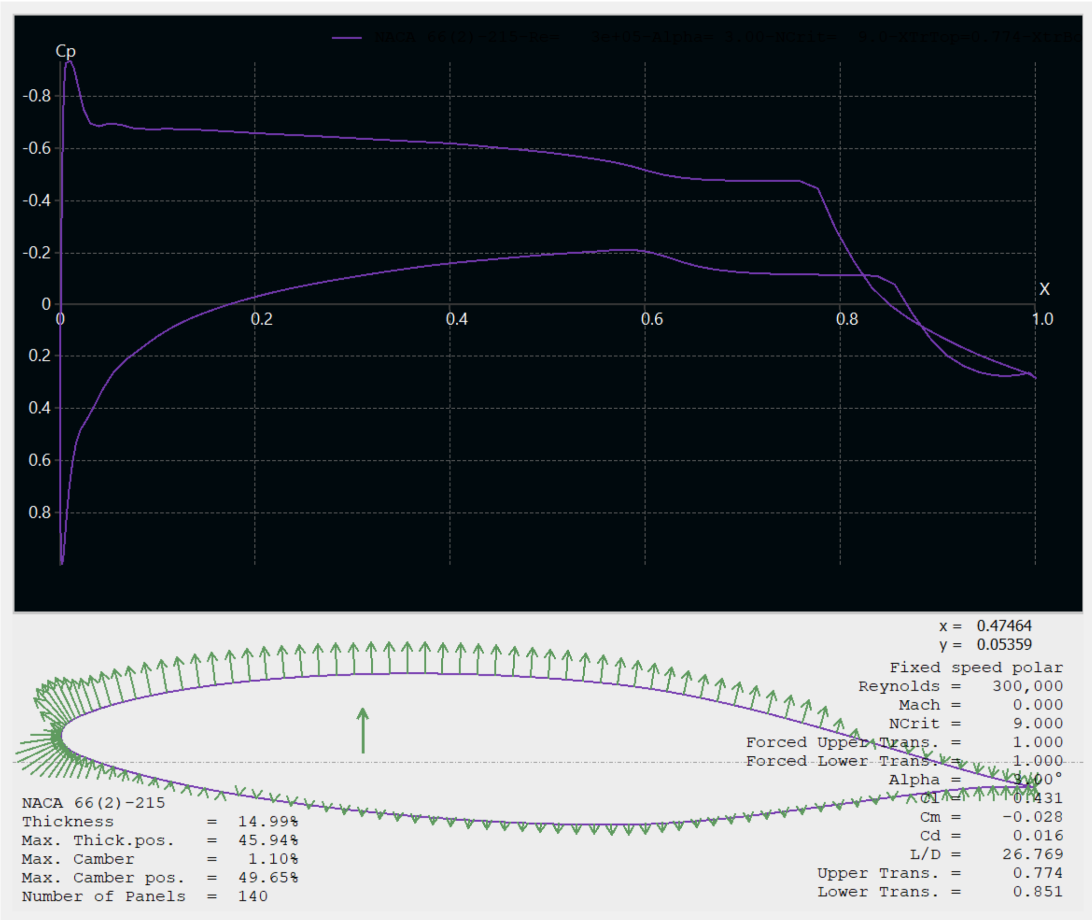


Figure 2.5: Pressure vectors and surface pressure coefficient distribution at 5 degrees angle of attack obtained using XFOIL/XFLR5.

2.4.5 Calculating Lift and Drag from the Pressure Distribution

These notes provide a summary of how one calculates the lift and drag from a pressure distribution. For complete details, see Fundamentals of Aerodynamics by Anderson, Section 1.5.

The pressure and shear stress distribution on an airfoil produce a resultant force and moment. This force and moment can be decomposed into components normal and parallel to the chord or freestream, as shown in Figure 2.6.

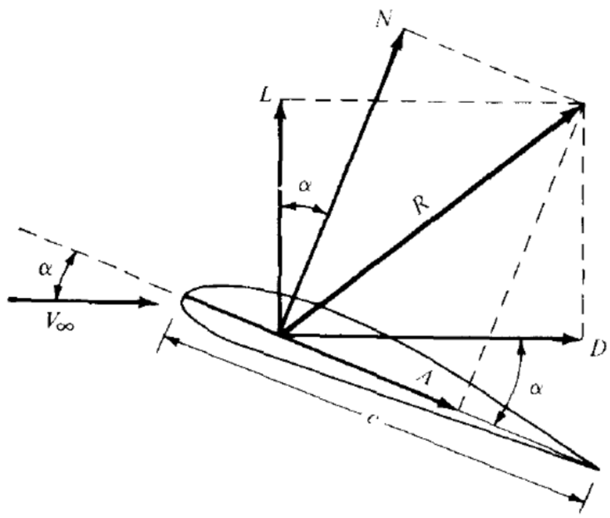


Figure 2.6: Resultant aerodynamic force and its components normal and parallel to the airfoil chord and freestream

The force component normal to the airfoil chord is denoted N, and the force component parallel to the airfoil chord is denoted A; the units are force per unit span since the airfoil is 2D. The force coefficients are then defined as in equation 2.1 and 2.2.

$$C_n = \frac{N}{q_\infty c} \tag{2.1}$$

$$C_a = \frac{A}{q_\infty c} \tag{2.2}$$

where $q_\infty = \frac{1}{2}\rho V_\infty^2$ is the dynamic pressure and c is the chord length.

The normal and axial force coefficients can be calculated by integrating the pressure and skin friction distributions over the airfoil surface as shown in Equations 2.3 and 2.4:

$$C_n = \frac{1}{c}\left[\int_0^c (C_{p,l} - C_{p,u})\,dx + \int_0^c \left(C_{f,u}\frac{dy_u}{dx} + C_{f,l}\frac{dy_l}{dx}\right)dx\right] \tag{2.3}$$

$$C_a = \frac{1}{c}\left[\int_0^c \left(C_{p,u}\frac{dy_u}{dx} - C_{p,l}\frac{dy_l}{dx}\right)dx + \int_0^c (C_{f,u} + C_{f,l})\,dx\right] \tag{2.4}$$

where C_p and C_f are the pressure and skin friction coefficients, respectively, defined as:

$$C_p = \frac{p - p_\infty}{q_\infty} \tag{2.5}$$

$$C_f = \frac{\tau}{q_\infty} \tag{2.6}$$

The term dy/dx is the local slope of the airfoil surface where the pressure is being measured and the subscripts u and l denote the upper and lower surfaces of the airfoil, respectively.

Note that we won't be calculating the skin friction drag in this experiment. It is assumed to be zero for the purposes of this lab. The subscripts l and u denote the airfoil lower and upper surfaces, respectively, and $\frac{dy}{dx}$ is the local slope of the airfoil surface.

Once C_n and C_a are obtained, the coefficients of lift and drag can be calculated:

$$C_l = C_n cos\alpha - C_a sin\alpha \tag{2.7}$$

$$C_d = C_n sin\alpha + C_a cos\alpha \tag{2.8}$$

2.5 Recommended Reading

It is highly recommended that students read the following sections in Fundamentals of Aerodynamics by John D. Anderson:

- 1.5 Aerodynamic Forces and Moments
- 3.5 Pressure Coefficient
- Chapter 4 Preview Box

2.6 Data to be Acquired

1. Dynamic pressure inside the wind tunnel during operation
2. Pressure readings from various pressure taps
3. Dimensions of airfoils used (for obtaining the coefficients)

2.7 Lab Setup and Procedure

2.7.1 Experimental Setup and Preparation

Lab Location: ARMS B098, Low Subsonic Wind Tunnel
In this experiment the pressure distribution over a laminar flow airfoil will be measured. The pressures will be recorded on the computer using a LabVIEW program. The wind tunnel must be off when the program is started so the readings are properly "zeroed." Once the LabVIEW program is opened and running, the wind tunnel can be turned on and the experiment performed.

Two pressure scanner systems, labeled "A" and "B", are used to acquire pressure data from the airfoil surfaces. Each scanner has 16 ports, labeled A1-A16 and B1-B16. Each port is connected to a piezoresistive pressure transducer that measures the differential pressure between the pressure at the port and a reference pressure. In this lab, the reference pressure port (labeled "RUN REF" on the scanner) is connected to the side of the wind tunnel so the reference pressure is the freestream static pressure p_∞. The pressure scanners are connected to the lab computer and LabVIEW is used to read the data from the 32 channels simultaneously. A pressure tap panel provides a convenient mechanism for connecting the pressure scanner ports to the pressure taps for a given test article. The NACA $66_2 - 215$ airfoil model used in this lab has 26 pressure taps. Taps 1-19 are on the top surface of the airfoil and are numbered in ascending order from the leading edge (LE) to the trailing edge (TE). Taps 20-26 are on the bottom surface of the airfoil and are also numbered in ascending order from the LE to the TE. The pressure tap numbers, x/c locations of the taps, and the associated pressure scanner port numbers are given in Table 2.1.

Airfoil Surface	Tap Number	x/c	Pressure Scanner	Channel Number
Upper Surface	1	0.0125	**Pressure Scanner Box A**	A1
	2	0.03125		A2
	3	0.0625		A3
	4	0.09375		A4
	5	0.125		A5
	6	0.1875		A6
	7	0.25		A7
	8	0.3125		A8
	9	0.375		A9
	10	0.4375		A10
	11	0.5		A11
	12	0.5625		A12
	13	0.625		A13
	14	0.6875		A14
	15	0.71875		A15
	16	0.75		A16
	17	0.78125	**Pressure Scanner Box B**	B1
	18	0.8125		B2
	19	0.875		B3
Lower Surface	20	0.0375		B4
	21	0.125		B5
	22	0.25		B6
	23	0.375		B7
	24	0.5		B8
	25	0.625		B9
	26	0.75		B10
Unused				B11
				B12
				B13
				B14
				B15
				B16

Table 2.1: Pressure tap numbers and x/c locations for the NACA $66_2 - 215$ airfoil model and the associated pressure scanner channel numbers.

Preparation

1. One member should log into the lab computer using their career account and navigate to the Lab 2 folder at the destination: `C:\ Temp \ AAE33401 \ Lab 2 - Pressure Tap Airfoil`. Make sure you save any and all data generated during the lab in the `data_33401_lab2` folder. Make a folder with your team name (e.g. Black16) within that folder.

2. Download the Lab 2 data spreadsheet from Brightspace, and open it on a computer. The spreadsheet will help you record data and perform calculations.

3. Open the LabVIEW VI LLB (virtual instrument LabVIEW Library)" **Lab2_33401_PressureTapAirfoil**" and then open "Pressure Tap Airfoil Measurement.vi." Click here to learn more about LabVIEW. **Please do not alter or modify any LabVIEW VIs - even very small changes can cause the program to stop working!**

4. Set the file path in the LabVIEW VI to your data folder. After the path, add a backslash and enter a name for the data file (in .txt format). For example, your final path might look like: `C:\temp\AAE33401\Lab 2 - Pressure Tap Airfoil\data_33401_lab2\Black16\ Black16_airfoil_data.txt`.

5. The airfoil model (NACA $66_2 - 215$) has 19 taps on the top surface and 7 taps on the bottom surface. Verify that the taps 1 through 19 are connected to channels A1 through B3 on the pressure tap panel and taps 20 through 26 are connected to channels B4 to B10.

2.7.2 Experimental Procedure

1. Run the **VI WHILE THE TUNNEL IS OFF** so it zeroes out the pressure channels.
   - The program will take a few seconds to begin running, so please wait.
   - The program will be ready to take data when you see the readings from the pressure scanner channels fluctuating around zero.

2. Set the airfoil angle of attack to -8 degrees and tighten down the wing nuts so the airfoil is secure.

3. Turn on the wind tunnel and set the speed to 30 Hz. Record the dynamic pressure to calculate the free stream velocity, in "Free Stream Velocity" worksheet in the Lab 2 Data spreadsheet.

4. Observe the pressure plot in LabVIEW the on the upper and lower surfaces of the airfoil to ensure that it looks as expected. **Show the pressure plot to your TA and explain why or why not it looks as expected.**
   - Note that the LabVIEW program reads and plots values of the differential pressure $p - p_\infty$ in units of psi.

5. Click the "Push to Write" button to record the pressure readings to the text file you defined in Step 4.

6. Turn off the wind tunnel and open the text file.
   - The first line in the data file contains the locations of the pressure taps, x/c.
   - The second line is the data you just recorded, with the angle of attack in the first column and followed by the pressure readings (tap 1 in column 2, tap 2 in column 3, etc.).

7. **Data Analysis - Pressure Coefficient** This part of the procedure will walk you through the steps to analyze the pressure data to obtain the pressure coefficient, c_p vs. x/c.

   7.1 Copy the x/c and pressure values to the first two columns of the "AOA = -8 deg" worksheet in the Lab 2 spreadsheet. Convert the pressure to Pa in the third column.

   7.2 Divide the differential pressures (in Pa) by the dynamic pressure you recorded from the pitot probe to obtain the pressure coefficients. Plot C_p vs. x/c and label the top and bottom airfoil surfaces on the plot. Do your results make sense? **Show your plot to the TA.**

8. **Data Analysis - Estimating Lift and Drag** This part of the procedure will walk you through the steps to estimate normal and axial forces and then the lift and drag from your experimental data.

   The normal and axial force coefficients can be calculated by integrating the pressure and skin friction distributions over the airfoil surface as shown in Equations 2.3 and 2.4.

In this lab, we only measure the surface pressure and do not measure the shear stress (which is challenging to do in practice). Hence, the integrals involving the skin friction coefficient C_f are neglected to obtain simplified equations for C_n and C_a:

$$C_n = \frac{1}{c}\int_0^c (C_{p,l} - C_{p,u})\, dx \tag{2.9}$$

$$C_a = \frac{1}{c}\int_0^c \left(C_{p,u}\frac{dy_u}{dx} - C_{p,l}\frac{dy_l}{dx}\right) dx \tag{2.10}$$

(Note: The skin friction becomes prominent when where there are large gradients in the flow velocity at the surface, which is usually a feature of turbulence. By using a laminar airfoil like the NACA $66_2 - 215$ we delay or entirely prevent laminar-to-turbulent transition of the boundary layer. Thus, ignoring skin friction effect will still yield reasonable results for the force coefficients).

We do not have continuous functions for the pressure coefficient but rather measurements at discrete locations. Therefore, we have to integrate Equations 2.9 and 2.10 numerically. A simple numerical integration method is the Trapezoidal Rule for a non-uniform grid (spacing between pressure measurement locations in our case). Using the Trapezoidal Rule to discretize the integrals gives:

$$\begin{aligned} C_n &= \frac{1}{c}\int_0^c (C_{p,l} - C_{p,u})\, dx = \frac{1}{c}\int_0^c C_{p,l} dx - \frac{1}{c}\int_0^c C_{p,u} dx \\ &\approx \left(\sum_{j=1}^{n} \frac{C_{p,j-1} + C_{p,j}}{2} \Delta\left(\frac{x}{c}\right)_j\right)_l - \left(\sum_{i=1}^{m} \frac{C_{p,i-1} + C_{p,i}}{2} \Delta\left(\frac{x}{c}\right)_i\right)_u \end{aligned}$$

$$\begin{aligned} C_a &= \frac{1}{c}\int_0^c \left(C_{p,u}\frac{dy_u}{dx} - C_{p,l}\frac{dy_l}{dx} dx\right) = \frac{1}{c}\int_0^c C_{p,u}\frac{dy_u}{dx} dx - \frac{1}{c}\int_0^c C_{p,l}\frac{dy_l}{dx} dx \\ &\approx \sum_{i=1}^{n} \left(\frac{C_{(p,i-1)}\left[\frac{\Delta\left(\frac{y}{c}\right)}{\Delta\left(\frac{x}{c}\right)}\right]_{i-1} + C_{(p,i)}\left[\frac{\Delta\left(\frac{y}{c}\right)}{\Delta\left(\frac{x}{c}\right)}\right]_i}{2} \Delta\left(\frac{x}{c}\right)_i \right)_u \\ &\quad - \sum_{j=1}^{m} \left(\frac{C_{(p,j-1)}\left[\frac{\Delta\left(\frac{y}{c}\right)}{\Delta\left(\frac{x}{c}\right)}\right]_{j-1} + C_{(p,j)}\left[\frac{\Delta\left(\frac{y}{c}\right)}{\Delta\left(\frac{x}{c}\right)}\right]_j}{2} \Delta\left(\frac{x}{c}\right)_j \right)_l \end{aligned}$$

Here $m = 19$ and $n = 7$ are the total number of taps on the upper and lower surface, respectively, and $\Delta(x/c)$ is the difference in x/c between two consecutive taps, while $\left[\frac{\Delta\left(\frac{y}{c}\right)}{\Delta\left(\frac{x}{c}\right)}\right]$ is the local slope of the airfoil surface at a particular tap. The local surface slope can be estimated from the airfoil surface coordinates provided as a .DAT file on Brightspace (also given in Section 2.8.2). An example calculation of the local surface slope is given in Section 2.8.1. After calculating the normal and axial force coefficients, the lift and drag coefficient can be found using Equations 2.7 and 2.8.

8.1 ☐ In the appropriate column in the data spreadsheet, calculate $\Delta(x/c)$, C_n terms, Local slope (a "Worksheet for slope calculation" has been provided for your reference) and C_a terms.

8.2 ☐ Calculate the final C_n, C_a and then C_l and C_d. **Show your calculated values for C_l and C_d at α = -8 degrees to your TA.**

9. ☐ Repeat the measurements for the angle of attack increments in the data spreadsheet, until after stall is achieved ($\sim$ 18 degrees). Record the data in the data spreadsheet and compile the C_l and C_d data.

10. ☐ **Return the experiment and the lab to its state when you first arrived.** Return all tools and equipment back to their designated location, turn off any equipment that was started for the purpose of your lab(you may leave the PC on, and sign out of your account instead), and clean up your workspace. **Obtain approval from the TA before leaving the lab.** Thank you!

2.8 Additional Information

2.8.1 Example Calculations: $\Delta(x/c)$ and $\Delta(y/c)$

Based on the calculation process outlined in the Experimental Procedure, consider an example calculation to understand how the slope is calculated at various pressure taps of the airfoil.

Suppose we need to calculate $\Delta(x/c)$, $\Delta(y/c)$ at the first tap location, which is located at $x/c = 0.0125$. Using the airfoil coordinates provided in the previous section, identify the two coordinates with values of x/c that are right before and right after 0.0125. We then perform the following calculations to estimate $\Delta(x/c)$ and $\Delta(y/c)$ as follows.

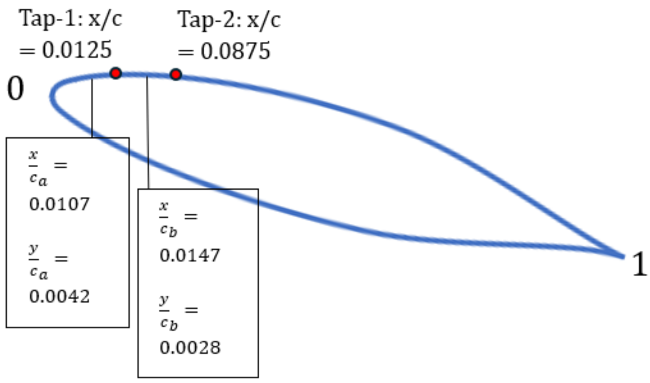


Figure 2.7: Image for example calculation for $\Delta(x/c)$, $\Delta(y/c)$

Calculation of $\Delta(x/c)$

From figure 2.7, value of $\Delta(x/c)$ for tap 1 is given by

$$\Delta\left(\frac{x}{c}\right)_1 = 0.0125 - 0$$

Consequently, for tap 2, it can be written as,

$$\Delta\left(\frac{x}{c}\right)_2 = 0.0875 - 0.0125$$

and so on. Thus for tap 19, one can write an expression

$$\Delta\left(\frac{x}{c}\right)_{19} = \Delta\left(\frac{x}{c}\right)_{18} - \Delta\left(\frac{x}{c}\right)_{19}$$

Calculation of $\Delta(y/c)$

For calculating $\Delta(y/c)$ at a specific tap location, one needs to look up the dat file data, and find the nearest x/c location, before and after the given tap location. For tap 1, those locations are boxed in the figure 2.7 and they are derived from table in section 2.8.2.

$$\left[\frac{\Delta\left(\frac{y}{c}\right)}{\Delta\left(\frac{x}{c}\right)}\right]_1 = \frac{0.0042 - 0.0028}{0.0142 - 0.0107} = \frac{0.0014}{0.0035} = 0.4$$

These values are entered in the "delta (x/c)" and "Local Slope" columns of the Lab 2 data spreadsheet, and the calculation is repeated for all surface taps.

2.8.2 NACA 66_2-215irfoil Coordinates

The coordinates for the NACA 66_2-215 surface (y/c) vs. x/c are given in the table below and are provided in a DAT file on Brightspace.

Sl No	x/c	y/c	Sl No	x/c	y/c	Sl No	x/c	y/c
1	1	0	61	0.018428	0.021736	121	0.69066	-0.04741
2	0.993359	0.001014	62	0.014154	0.019493	122	0.708565	-0.04449
3	0.982368	0.002802	63	0.010714	0.017364	123	0.726592	-0.0414
4	0.969897	0.004996	64	0.007887	0.015289	124	0.744778	-0.03816
5	0.955711	0.007707	65	0.005552	0.013257	125	0.763139	-0.0348
6	0.939801	0.011019	66	0.003664	0.011195	126	0.781659	-0.03133
7	0.922598	0.014866	67	0.00221	0.00905	127	0.80026	-0.02781
8	0.904739	0.019047	68	0.001146	0.006896	128	0.818806	-0.02428
9	0.886614	0.023373	69	0.000429	0.00477	129	0.8372	-0.02081
10	0.868296	0.027778	70	0.000035	0.002683	130	0.855455	-0.01743
11	0.849849	0.032225	71	-4.3E-05	0.000635	131	0.873648	-0.01416
12	0.831394	0.036664	72	0.000211	-0.00139	132	0.891734	-0.01102
13	0.813042	0.041041	73	0.000798	-0.0034	133	0.909485	-0.00805
14	0.794835	0.045322	74	0.001723	-0.00542	134	0.926545	-0.00541
15	0.776758	0.049485	75	0.00301	-0.00745	135	0.942539	-0.00329
16	0.758795	0.053512	76	0.004698	-0.00948	136	0.957211	-0.00178
17	0.740942	0.05738	77	0.00682	-0.01142	137	0.97049	-0.0008
18	0.723207	0.061065	78	0.009398	-0.01326	138	0.982471	-0.00025
19	0.705579	0.06455	79	0.012501	-0.01505	139	0.993316	-1.9E-05
20	0.688027	0.067819	80	0.01626	-0.01683	140	1	0
21	0.670531	0.070857	81	0.020916	-0.01864			
22	0.653132	0.073641	82	0.02688	-0.02064			
23	0.635927	0.076136	83	0.03463	-0.02306			
24	0.618971	0.078307	84	0.044343	-0.0259			
25	0.602161	0.080134	85	0.055547	-0.02884			
26	0.585253	0.081615	86	0.06837	-0.03181			
27	0.567995	0.082808	87	0.083277	-0.03492			
28	0.550302	0.083776	88	0.099487	-0.03801			
29	0.53232	0.084567	89	0.116098	-0.04088			
30	0.514272	0.085177	90	0.133076	-0.04352			
31	0.496252	0.085604	91	0.150445	-0.04597			
32	0.478219	0.085851	92	0.168115	-0.04824			
33	0.460119	0.085929	93	0.185952	-0.05033			
34	0.441966	0.085845	94	0.203875	-0.05224			
35	0.423829	0.085602	95	0.221872	-0.05398			
36	0.405746	0.085192	96	0.239951	-0.05557			
37	0.3877	0.084607	97	0.258113	-0.057			
38	0.369644	0.083846	98	0.276344	-0.0583			
39	0.351559	0.082915	99	0.294624	-0.05946			
40	0.333465	0.08182	100	0.312936	-0.06049			
41	0.315401	0.080559	101	0.331261	-0.06139			
42	0.297391	0.07913	102	0.349586	-0.06216			
43	0.279442	0.077529	103	0.367902	-0.06279			
44	0.261556	0.07575	104	0.386211	-0.06329			
45	0.243738	0.073785	105	0.404526	-0.06366			
46	0.225995	0.07163	106	0.422872	-0.06389			
47	0.208338	0.069278	107	0.441252	-0.064			
48	0.190784	0.066724	108	0.459632	-0.06397			
49	0.173362	0.063959	109	0.477955	-0.06383			
50	0.156126	0.060971	110	0.496216	-0.06355			
51	0.139162	0.057759	111	0.514467	-0.06312			
52	0.122556	0.054319	112	0.532745	-0.06255			
53	0.106346	0.050636	113	0.550942	-0.06183			
54	0.090516	0.046688	114	0.568822	-0.06096			
55	0.075332	0.042547	115	0.586254	-0.05989			
56	0.061762	0.038516	116	0.603353	-0.05855			
57	0.050087	0.034671	117	0.620396	-0.05689			
58	0.03979	0.030875	118	0.637628	-0.05491			
59	0.030933	0.027296	119	0.655134	-0.05265			
60	0.023885	0.024252	120	0.672842	-0.05014			

Table 2.2: **DAT file data for NACA** $66_2 - 215$

Chapter 3

Finite Wings

3.1 Introduction

Wings and airfoils are both fundamental to the science of aerodynamics, but they serve distinct roles in aircraft design. An airfoil refers to the cross-sectional shape of any surface designed to generate lift, such as the shape of a wing, propeller blade, or rotor blade. Its purpose is to optimize airflow and create lift by manipulating the pressure difference between the upper and lower surfaces. In contrast, a wing is the full three-dimensional structure of an aircraft component designed for flight, which incorporates the airfoil shape along its span. Wings take the basic principles of airfoils and apply them across a broader scale, incorporating factors like span, aspect ratio, and control surfaces that contribute to an aircraft's performance. Span, or the length of the wing from tip to tip, has a significant effect on how a wing performs. A longer span generally improves lift generation and reduces induced drag, leading to more efficient flight, especially at low speeds. However, longer wings can increase structural weight and reduce maneuverability. To mitigate induced drag, which arises due to pressure differences at the wing tips, engineers often incorporate wingtip devices such as winglets, raked tips, or blended wingtips. These devices help to smooth out airflow and reduce vortices, improving fuel efficiency and overall aerodynamic performance. By understanding the interactions between airfoils, wing span, and wingtip devices, designers can optimize aircraft for a range of performance metrics, from speed to fuel efficiency.

3.2 Lab Overview

In Labs 1 and 2, we studied the lift, drag, and surface pressure distributions of airfoils, using experimental designs that created mostly two-dimensional flow fields and thus the spanwise direction was not considered. In this lab, wing models with finite span will be used, allowing fully three-dimensional flow and wing tip effects. The lift and drag will be measured for five different wing configurations: three rectangular wings with varying aspect ratio and a small aspect ratio wing with small and large wingtip devices attached. The effect of aspect ratio on the wing performance will be investigated and the effects of adding winglets vs. increasing span will be compared. The experiments will be conducted in the large test section Boeing Wind Tunnel at Purdue's Aerospace Sciences Laboratory. The 6 ft. x 4 ft. test section allows for testing large aspect ratio wings without significance tunnel wall interference.

3.3 Objectives

During this experiment, you will:

1. Be introduced to the effect of finite span on wing aerodynamics.
2. Make measurements of finite-span effects for 3 span/chord ratios.
3. Study the effect of winglets and compare them to changing the aspect ratio.

3.4 Background

3.4.1 Finite Wing Theory

In two dimensions, point vortices can be used to describe a flow about an airfoil with circulation. This does not suffice for a three-dimensional model: the point vortex must be replaced by a vortex filament consisting of a bound vortex fixed in the wing (the lifting line), trailing vortices extending downstream, and the starting vortex shed when the motion is initiated. As no vortex element terminates in the fluid, Helmholtz's vortex theorem has not been violated. If the circulation distribution is uniform across the span of the wing, the trailing vortices will be shed only where the circulation changes, at the wingtips. If the circulation distribution varies along the span the result will be a continuous distribution of vortex filaments trailing downstream, the strength of any filament being related to the change of the circulation at that span wise section. For a discrete representation this is the well-known horseshoe vortex system (Figure 3.1).

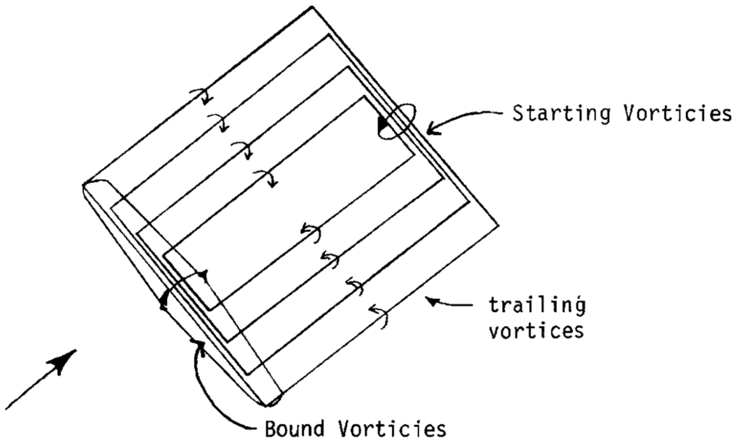


Figure 3.1: Vortex representation for a finite wing

We can observe the physics of vortex formation by noting that the higher pressure existing below the wing and the reduced pressure above cause the air flow to roll around the tips; in fact, the incoming flow passing over and beneath the wing receives on the lower side a velocity component outward and on the upper side a component inward.

An upward-lifting wing must on average have a pressure difference between its upper and lower surfaces. Outside of the wingtips, this pressure difference must disappear. Consequently, fluid particles which approach the wingtip above the wing are subjected to a spanwise pressure gradient which causes them to curve towards the wing center. Fluid particles which approach below the wing will curve away from the wing center. Particles close to the tip will tend to flow around the tip edge, from the lower to the upper surface (Figure 3.2).

This relative velocity or discontinuity of the two flows at the trailing edge is exactly the result of the surface distribution of vortices mentioned before; the vortex filaments trailing downstream can be considered appropriately a vortex sheet rolling up behind and slightly inside of the wingtips. (Figure 3.3).

After a time the starting vortex is sufficiently far downstream to allow changes in the flow field about the wing to be due solely to the bound and trailing vortices. The downward component of velocity induced perpendicular to the direction of motion is called the downwash, w, and has the effect of altering the angle of attack as seen by the wing. An understanding of the induced flows due to a finite and an infinite wing can be studied qualitatively. Figure 3.4 shows the change of the vertical velocity component along a line parallel to the direction of flight for an infinite wing - the rising flow in front of the wing, the descending flow behind the wing, and the resulting lift vector.

For the finite wing, the downward velocity due to the trailing vortices must be added to the two-dimensional situation (Figure 3.5). The velocity $\vec{V}$ seen by the wing is the vector sum of the freestream velocity $\vec{V}_\infty$ and the downwash $\vec{w}$ at the wing. The "lifting" force, i.e. that force perpendicular to the velocity AS SEEN BY THE WING, is now rotated; the induced velocity w can be considered as a reduction of the rising flow at the leading edge due to the deviation from an infinitely long wing. At a great distance behind the wing the descending velocity is 2W, which demonstrates that the velocity due to a straight vortex filament extending to infinity in

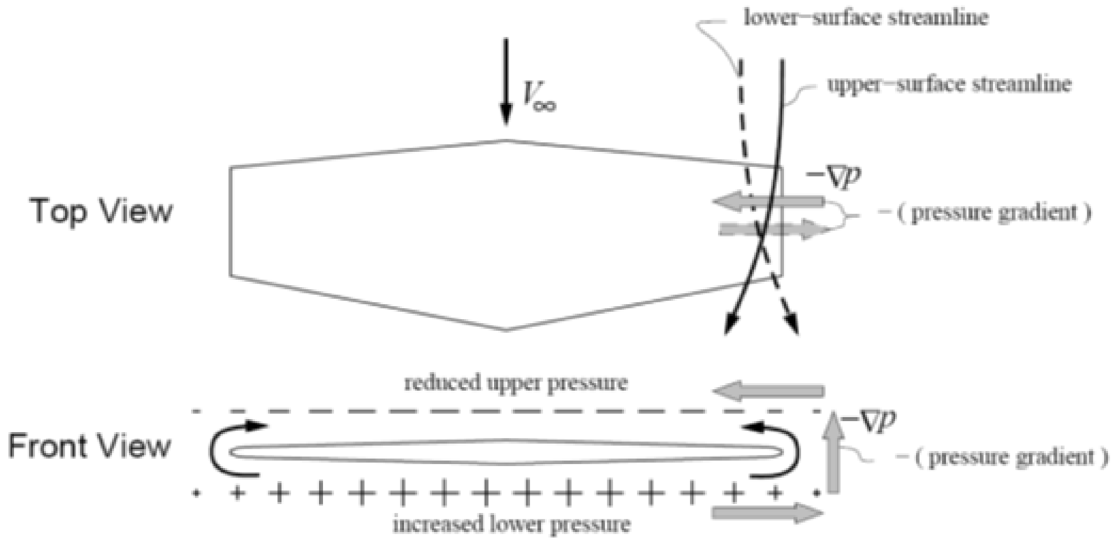


Figure 3.2: Spanwise flow on a finite wing.

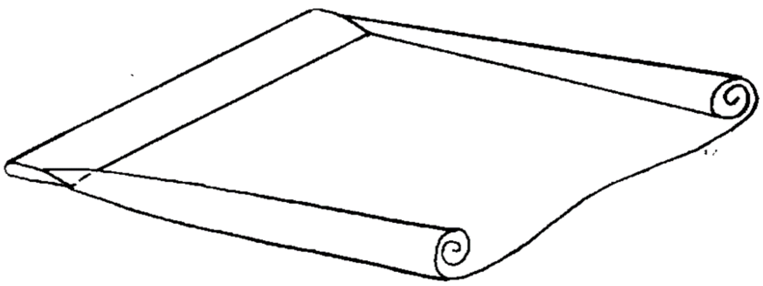


Figure 3.3: Vortex sheet shed from a finite wing

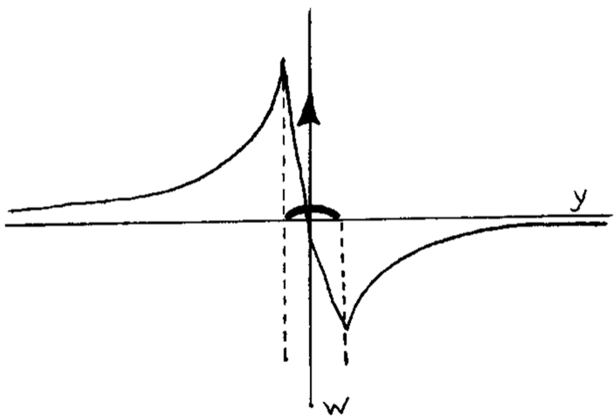


Figure 3.4: Vertical velocity near a 2D wing

both direction (such as a point in the wake far downstream) is twice that due to a filament extending to infinity in one direction only (as for a point on the wing itself).

The velocity field associated with the two trailing vortices is mostly downward (opposite to lift direction) directly behind the wing, and upward outside the tip vortices. The downwash can also be viewed as the result of the lifting wing "pushing down" on the air, which results in the air having added downward momentum directly behind the wing. (Figure 3.6).

The Kutta-Joukowski theorem states that when circulation exists in a velocity field, a force $\vec{F}$ is created: $\vec{F} = \rho\vec{V} \times \vec{\Gamma}$ perpendicular to $\vec{V}$ and to $\vec{\Gamma}$ (see Figure 3.7). Since lift and drag are defined as the components

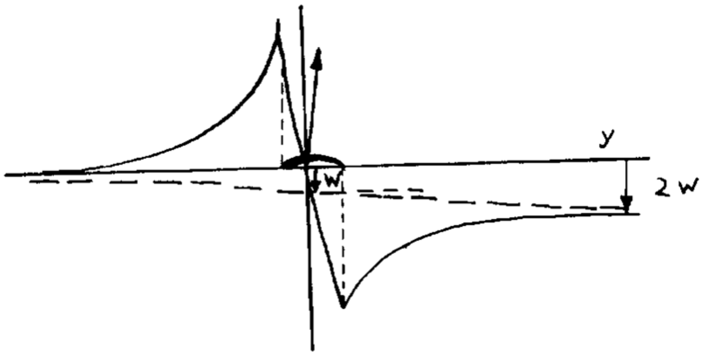


Figure 3.5: Vertical velocity near a finite wing

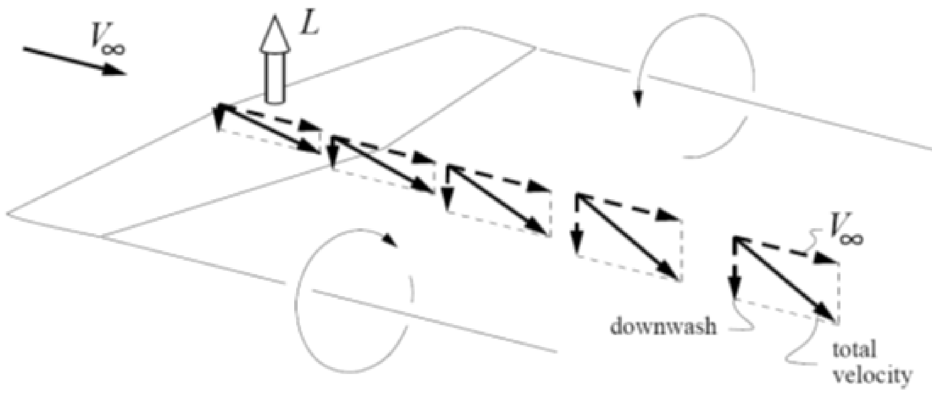


Figure 3.6: Downwash in the wake of a wing

of the resultant force perpendicular and parallel to the freestream velocity $\vec{V}_\infty$, now has a drag component, $D_i = -\rho \left|\vec{W}\right| \cdot \left|\vec{\Gamma}\right|$ called the induced drag, with a magnitude per unit span . Note that for the case of an infinite wing $\vec{w} = \vec{0}$; therefore $\vec{V} = \vec{V}_\infty$ and no induced drag appears.

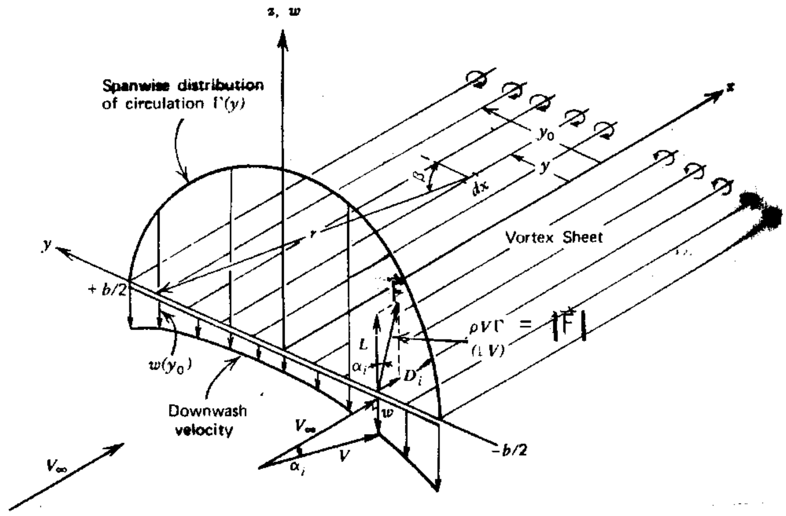


Figure 3.7: Circulation distribution for a finite wing

Three angles of attack can be considered with the problem shown in Figure 3.7:

1. $\alpha =$ "geometric" angle of attack measured from flight path V_∞ to chordline. (Not shown.)

2. α_i = induced angle of attack, measured from the relative wind (V) to the flight path (V_∞).
3. α_{Lo} = angle of zero lift, measured from the chord to the "Zero lift line". (Not shown.)

Glauert's found that the minimum value of $C_{D,i}$ occurs when the downwash is a constant, which yields an elliptical distribution of lift across the span. Expression for coefficient of induced drag is given by equation 3.1

$$C_{D,i} = \frac{C_L^2}{\pi \mathrm{AR}} \tag{3.1}$$

where $AR = b^2/s$ which is referred to as the aspect ratio. Corrections can be made for other than elliptic distributions, to a first-order approximation, by writing the equation as shown in equation 3.2.

$$C_{D,i} = \frac{C_L^2}{\mathrm{e}\pi \mathrm{AR}} \tag{3.2}$$

where e is the efficiency factor.

Experimental investigation has shown that if the aspect ratio is not too small, the lift distribution across the span for a rectangular wing does not deviate greatly from the elliptic type.

Thus we can write the total drag coefficient as a sum of the profile drag and the induced drag:

$$C_D = C_{D,o} + C_{D,i} \tag{3.3}$$

The theory does not predict stall or profile drag.

A last comment can be made upon the distribution of lift, downwash, and induced drag upon an elongated wing (Figure 3.8). The transition to an infinite wing is apparent - constant lift, zero downwash, and zero induced drag.

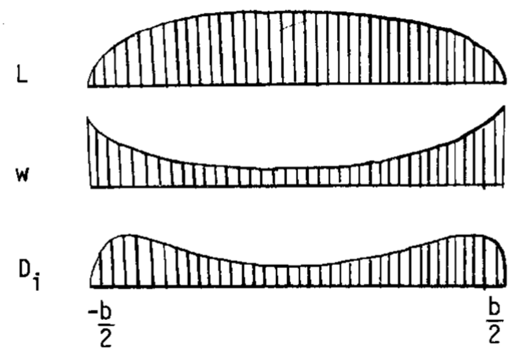


Figure 3.8: Spanwise loading distribution for a finite wing

3.4.2 Experimentally Observed Results

The drag polar, or C_L versus C_D, is a useful indication of the variation of drag with lift, and gives the important characteristics for a particular wing as to what drag penalties must be suffered for a required value of C_L. Figure 3.9 shows an example. The slope of the line joining the origin ($C_D = 0$, $C_L = 0$) to the tangent of the polar gives $(L/D)_{max}$ and at what value of C_L and C_D this is achieved. Such results are helpful when determining the applicability of certain wing sections to the preferred operating conditions. Figure 3.9 also shows theoretical drag, which predicts only the lift-induced component, compared to a typical polar diagram for observed drag. The difference between the two depends upon the shape of the wing profile and has been shown to be independent of aspect ratio; hence called "profile drag".

Figure 3.10 shows another drag polar together with a lift-curve slope. The effect of aspect ratio upon the lift curve and drag polar is marked, which follows from what we have noted previously; C_L varies due to the change in effective angle of attack with aspect ratio, and C_D varies due to a direct aspect ratio effect. For an untwisted wing, the angle of zero lift and the drag at zero lift can be seen to be independent of aspect ratio.

Figure 3.11 shows a few real life examples of vortices.

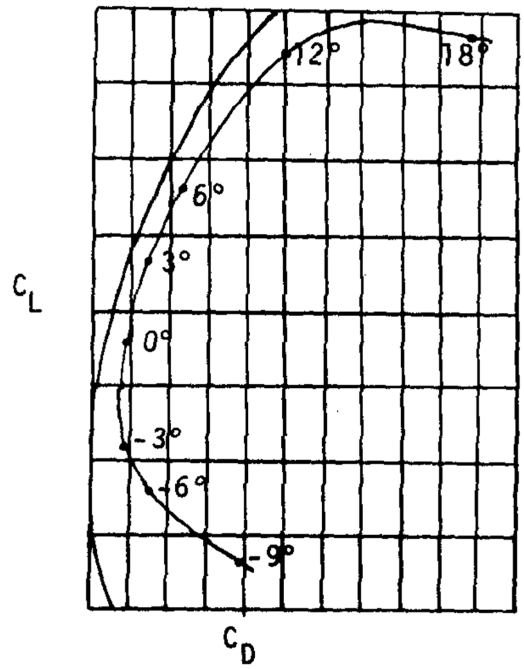


Figure 3.9: Sketch of a typical drag polar.

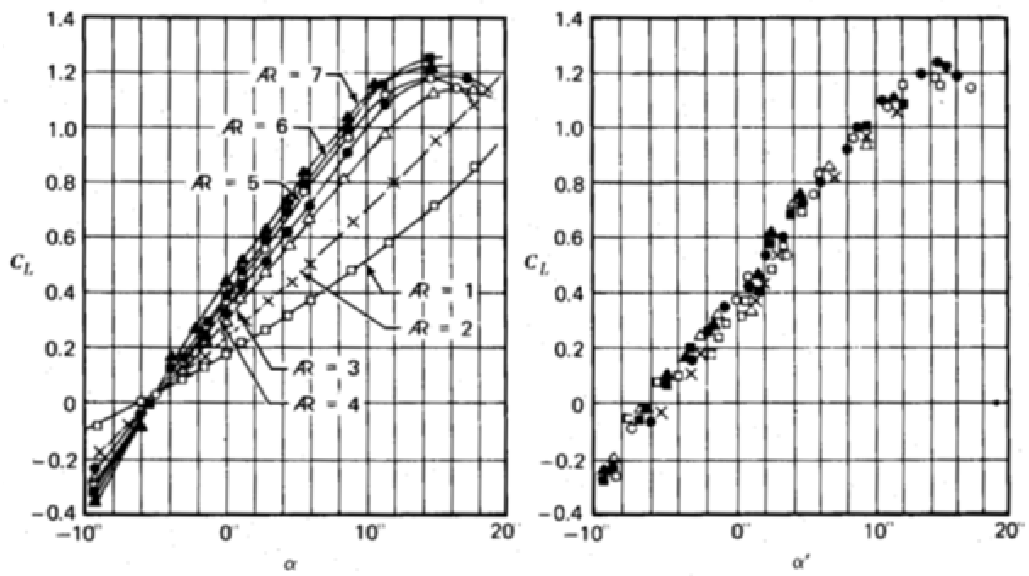


Figure 3.10: Lift curves for various aspect ratios.

3.5 Recommended Reading

It is highly recommended that students read the following sections in Fundamentals of Aerodynamics by John D. Anderson:

- 5.1 Introduction: Downwash and Induced Drag
- 5.3 Prandtl's Classical Lifting-Line Theory

3.6 Data to be Acquired

Following is the data that you need to acquire over the course of this lab:

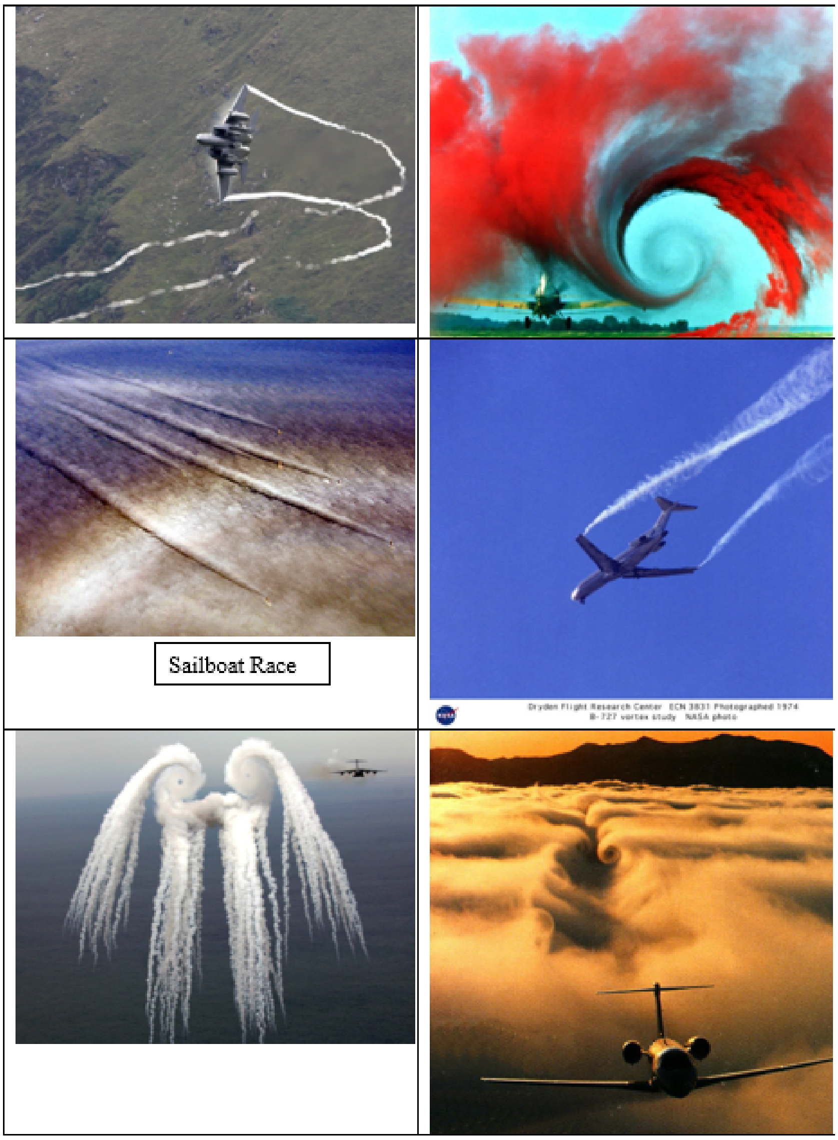


Figure 3.11: Flow Visualization Pictures of Wake Vorticies

1. Actual and measured weights used for calibration
2. Dimensions of rectangular wing models
3. Lift and Drag Forces for various angles of attacks and wing configurations

3.7 Lab Setup and Procedure

3.7.1 Experimental Setup and Preparation

Lab Location: ASL (Aerospace Sciences Lab), Boeing Wind Tunnel
In this lab you will measure the lift and drag for wings with 3 different aspect ratios and with 2 different wing tip devices. The configurations are listed as follows:

1. Base airfoil (NACA 0012, maximum thickness = 0.95 inches)
2. Base airfoil with small span additions
3. Base airfoil with larger span additions
4. Base airfoil with small wing tip devices
5. Base airfoil with large wing tip devices

Preparation

1. ☐ One member should log into the lab computer using their career account and navigate to the Lab 3 folder at the destination: `C:\ Temp \ AAE33401 \ Lab 3 - Finite Wings`. Make sure you save any and all data generated during the lab in the `data_33401_lab3` folder. Make a folder with your team name (e.g. Black16) within that folder.

2. ☐ Download the Lab 3 data spreadsheet from Brightspace, and open it on a computer. The spreadsheet will help you record data and perform calculations.

3. ☐ Open the LabVIEW VI "Lab3_33401_BoeingWindTunnel.vi." Click here to learn more about LabVIEW. Familiarize yourself with the VI: you will use it to change the angle of attack and record the flow speed and forces on the wing. **Please do not alter or modify any LabVIEW VIs - even very small changes can cause the program to stop working!**

4. ☐ Set the file path in the LabVIEW VI to your data folder. After the path, add a backslash and enter a name for the data file (in .txt format). For example, your final path might look like:
`C:\temp\AAE33401\Lab 3 - Finite Wings\data_33401_lab3\Black16\`
`Black16_data.txt`.

5. ☐ Click the Run arrow ⇨ at the top toolbar to begin running the VI and click the "TARE" button to zero the forces (lift and drag) and moments (pitch and roll). The readouts of the forces and moments should be fluctuating near zero. If the readouts are not fluctuating or there is an error message, get help from the TA.

6. Verify that the VI operates and saves the data correctly by completing the following steps:

   6.1 ☐ To change the angle of attack (AoA), enter the amount you want to change the angle by (in degrees) and click the "AoA" button. Try increasing the AoA by 2 degrees, then by another 2 degrees. Note that the value you enter in the "Relative AoA" field is only the relative change from the current angle. The absolute AoA is automatically updated on the VI.

   6.2 ☐ To return the wing to zero angle of attack, click the "Zero AoA" button.

   6.3 ☐ Click the "Push to Write" button in the VI to write the data to the file you specified in the LabVIEW path. A push is confirmed when you see the green "Success" light and when the push counter increases.

   6.4 ☐ Press the STOP button on the VI to stop the program.

   6.5 ☐ Open the data file using Notepad, Wordpad or Excel. The data file has 13 columns. The first row of data is the column headings ("Pitot", "Speed", "Temperature", "Wind Speed", "AoA", "Drag", "Lift", "Pitch", "Roll", "F0", "F1", "F2", "F3"). Note that F0-F3 are the raw force readings directly from the load cells in the force balance (see Section 7.2). The VI uses these values to calculate the forces (in lbf) and moments (lbf-ft) for you. The second row in the data file contains the readings corresponding to the headers in first row at the time you clicked the "Push to Write" button.

3.7.2 Experimental Procedure

1. **Calibrating the Force Balance**

   1.1 ☐ Make sure the tunnel is not running, contains no test article (apart from the base airfoil) in the test section and that there are no weights placed on the force balance.

   1.2 ☐ Click the "TARE" button to zero the forces and record the initial (zero actual weight) value of **lift** from the VI in the data spreadsheet.

   1.3 ☐ First, **calibrate the lift** by placing weights on the force balance top plate in the circular space marked "L". Apply known (total) weights of 0.5, 1.5, 2.5, 4.5, 6.5 and 11.5 lbs, starting with the 0.5 lb weight and stacking additional weights on top. Wait for the reading from the force balance to even out, then record the data from LabVIEW in Lab 3 data spreadsheet. Note that the curve is quite linear for the most part, but remember that calibration is imperative to get accurate data.

   1.4 ☐ Remove all weights from the force balance and click the "TARE" button. Record the initial (zero actual weight) value of **drag** from the VI in the data spreadsheet.

1.5 ☐ Calibrate the drag by placing weights on the hanging platform in the circular space marked "D". Apply known (total) weights of 0.5, 1.5, 2.5, 4.5, 6.5 and 11.5 lbs, starting with the 0.5 lb weight and stacking additional weights on top. Wait for the reading from the force balance to even out, then record the data from LabVIEW in Lab 3 data spreadsheet.

1.6 ☐ Plot the "Actual Weight" vs "Measured Weight" for both the lift and the drag and fit linear calibration curves. Record the values for the slopes and intercepts of the calibration curves in the appropriate cells in the worksheet. **Show your calibration curves to the TA before proceeding.**

2. ☐ Turn on the tunnel overhead lights using the flip switch above the lab computer.

3. ☐ Press the red button below the wind tunnel door to raise the door. Raise the door enough to comfortably climb in and out of the wind tunnel, please do not raise it all the way.

4. ☐ One team member should climb into the tunnel using a step ladder.

5. ☐ Measure the chord length and span of the base airfoil, which should already be mounted in the wind tunnel. Record the measurements in the data spreadsheet.

6. ☐ Make sure that the base wing is securely mounted and that there are no other loose items in the wind tunnel test section.

7. ☐ Exit the tunnel and press the green button to close the test section door.

8. ☐ Measure the spans of the small and large span additions and record in the data spreadsheet. You do **not** need to make any measurements for the wing tip devices.

9. ☐ Click "Zero AoA" in the VI to return the base wing to zero angle of attack (press "Zero AoA") and click the "TARE" button to zero the forces and moments. Leave the VI running.

10. ☐ Notify the TA that you are ready to turn on the tunnel. The TA will unlock the controls and verify that the tunnel drive frequency is set to 20 Hz.

11. ☐ At the instruction of the TA, press the small green button (labeled "Hand") on the controller to turn on the tunnel. Wait for the wind tunnel drive frequency to ramp up to 20 Hz and the dynamic pressure to settle to a constant value.

    Note: Always press the **small** red button (labeled "Off") on the controller to turn off the tunnel, this will allow the drive frequency to gradually decrease. **Only use the large Emergency Stop button in the case of a true emergency - this will immediately cut power to the drive which can cause damage!**

12. ☐ The LabVIEW VI displays and records the dynamic pressure from a Pitot tube ("Pitot Pressure" field) and the flow velocity measured using a hot film anemometer ("Anemometer Wind Speed" field). Use the Pitot tube reading to calculate the freestream velocity and compare to the anemometer reading. Record the dynamic pressure, wind speed calculated from the dynamic pressure, and wind speed from the anemometer in the data spreadsheet ("Dimensions and Pitot" worksheet).

    Note: The tunnel frequency will be kept at 20 Hz For all tests, **please do not increase the speed.**

13. **Measure lift and drag for a range of angle of attack.**

    13.1 ☐ Click the "Push to Write" button to record the zero AoA data to the file.

    13.2 ☐ Increase the angle of attack in increments of 2 degrees until you reach stall, writing the data at each increment. Note that the angle of attack will not increase past 20 degrees, but you should notice the onset of stall well before this point.

    13.3 ☐ Turn the tunnel off by pressing the small red "Off" button on the controller.

    13.4 ☐ Click "Zero AoA" to return the wing to zero angle of attack and then click the "STOP" button to stop the VI.

    13.5 ☐ Open your data file and copy the angle of attack, lift, and drag measurements to the Lab 3 data spreadsheet.

    13.6 ☐ Correct the lift and drag measurements using the calibration curve obtained in Step 1.6.

13.7 ☐ Calculate the lift and drag coefficients. Make two plots: 1) lift coefficient vs. angle of attack and 2) drag polar (drag coefficient vs. lift coefficient). **Show your plots to the TA before proceeding.**

14. ☐ Verify that the tunnel is off, then raise the door so one team member can install the ***small*** **span additions** to the base airfoil. Each span addition connects to the base airfoil using two mounting rods, there are no screws to tighten.

15. ☐ Verify that there are no other loose items in the wind tunnel test section, then exit the tunnel and close the door.

16. ☐ Verify that the wing is at zero angle of attack, click the "TARE" button, and turn on the tunnel.

17. ☐ Repeat Steps 13.1-13.5 to record the lift and drag for the base wing with small span additions.

18. ☐ Repeat Steps 14-17 with the ***large*** **span additions**. Make sure to remove the small span additions and install the large additions directly on the base wing.

19. ☐ Repeat Steps 14-17 with the ***small*** **wing tip devices**. Make sure to remove the all span additions and install the wing tip devices directly on the base wing.

20. ☐ Repeat Steps 14-17 with the ***large*** **wing tip devices**. Make sure to remove all span additions and install the wing tip devices directly on the base wing.

21. ☐ **Return the experiment and the lab to its state when you first arrived.** Return all tools and equipment back to their designated location, turn off any equipment that was started for the purpose of your lab, (you may leave the PC on, and sign out of your account instead) and clean up your workspace. **Obtain approval from the TA before leaving the lab.** Thank you!

Chapter 4

Supersonic Wind Tunnel

4.1 Introduction

Supersonic flow refers to the movement of a gas or fluid at a speed greater than the speed of sound in that medium. When an object travels at supersonic speeds, it moves faster than the pressure waves it generates, leading to the formation of shock waves. These shock waves create significant changes in the flow properties, such as pressure, temperature, and density, across their boundaries. Supersonic flows are characterized by Mach numbers greater than one ($M > 1$). The study of supersonic flow is essential in aerodynamics because it involves complex fluid dynamics, including shock waves and expansion fans, that do not appear in subsonic or transonic flow regimes.

Supersonic flow has various applications in real life, especially in the field of aerospace and defense. Supersonic aircraft, like fighter jets and experimental planes, use this principle to achieve high speeds, which are crucial for military and reconnaissance missions. Another prominent application is in the development of supersonic and hypersonic missiles, where the ability to reach targets quickly is a strategic advantage. In the commercial sector, research into supersonic travel aims to reduce flight times significantly, with projects like supersonic passenger jets under development.

A supersonic wind tunnel generates controlled, high-speed airflow with Mach numbers greater than one by accelerating high pressure gas through a converging-diverging nozzle. Such wind tunnels allow researchers to analyze aerodynamic forces, pressure distribution, and heat transfer on test models in a controlled environment.

4.2 Lab Overview

In this lab, we will learn about the operation of a supersonic wind tunnel and measure the pressure distribution along the nozzle and test section. We will also use an optical technique called schlieren visualization to observe the shock and expansion waves generated around different test models placed in the supersonic flow.

4.3 Objectives

In this experiment you will:

1. Measure the pressure distribution along the nozzle of a supersonic tunnel using a digital pressure transducer system.
2. Observe the shock waves and expansion fans using schlieren or shadowgraph visualization.
3. Measure the pressure across oblique shocks and expansion fans.
4. Measure the total pressure loss across a normal shock wave formed by a blunt body.

4.4 Background

4.4.1 Supersonic Flow through a Nozzle

Figure 4.1 shows the flow through a converging-diverging nozzle. Assume that the pressure difference between Region I and Region II is great enough that the flow in Region I is subsonic and the flow in Region II is

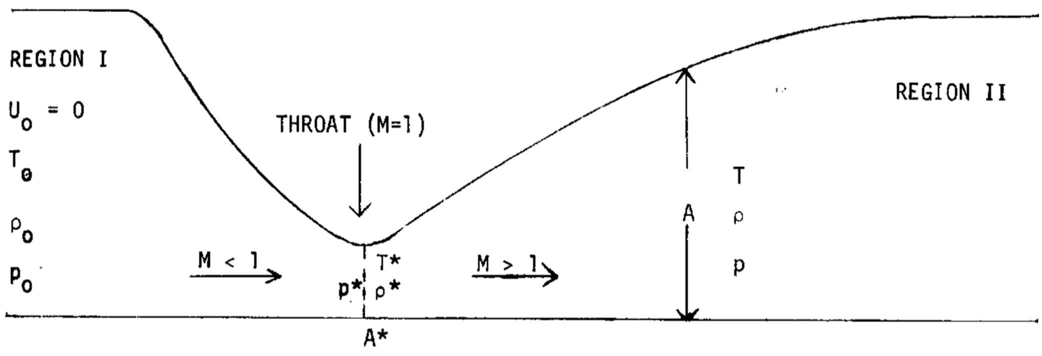


Figure 4.1: Supersonic Flow through a Nozzle

supersonic. Total quantities in the flow are designated by the subscript "o". The position along the nozzle with the smallest area is called the throat. Conditions where the flow is sonic (M = 1) are denoted with a "*" superscript.

From the continuity, momentum, and energy equations and the perfect gas laws the following relationships can be found:

$$\frac{P_o}{P} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{\frac{\gamma}{\gamma - 1}} \tag{4.1}$$

$$\frac{\rho_o}{\rho} = \left(1 + \frac{\gamma + 1}{2} M^2\right)^{\frac{1}{\gamma - 1}} \tag{4.2}$$

$$\frac{T_o}{T} = 1 + \frac{\gamma - 1}{2} M^2 \tag{4.3}$$

$$\left(\frac{A}{A^*}\right)^2 = \frac{1}{M^2}\left[\frac{2}{\gamma + 1}\left(1 + \frac{\gamma - 1}{2} M^2\right)\right]^{\frac{\gamma + 1}{\gamma - 1}} \tag{4.4}$$

where:
p = pressure at a particular point along nozzle
ρ = density at a particular point along nozzle
T = temperature at a particular point along nozzle
M = Mach number at a particular point along nozzle
A = area at a particular point along nozzle
γ = ratio of specific heats

These relationships are tabulated in the appendices of Fundamentals of Aerodynamics by John D. Anderson and other compressible flow textbooks.

4.4.2 Supersonic Flow Around Bodies

Supersonic Compression by Turning

When a sharp-edged body (wedge) is in supersonic flow an oblique shock wave is formed which is attached to the wedge (Figure 4.2a). For the attached shock wave the relationship between the shock angle (β), wedge angle (θ), and the Mach number ahead of the oblique shock wave (M_1) is:

$$\tan\theta = 2\cot\beta \frac{M_1^2 \sin^2\beta - 1}{M_1^2 (\gamma + \cos 2\beta) + 2} \tag{4.5}$$

$$M_2^2 \sin^2(\beta - \theta) = \frac{1 + \frac{\gamma - 1}{2} M_1^2 \sin^2\beta}{\gamma M_1^2 \sin^2\beta - \frac{\gamma - 1}{2}} \tag{4.6}$$

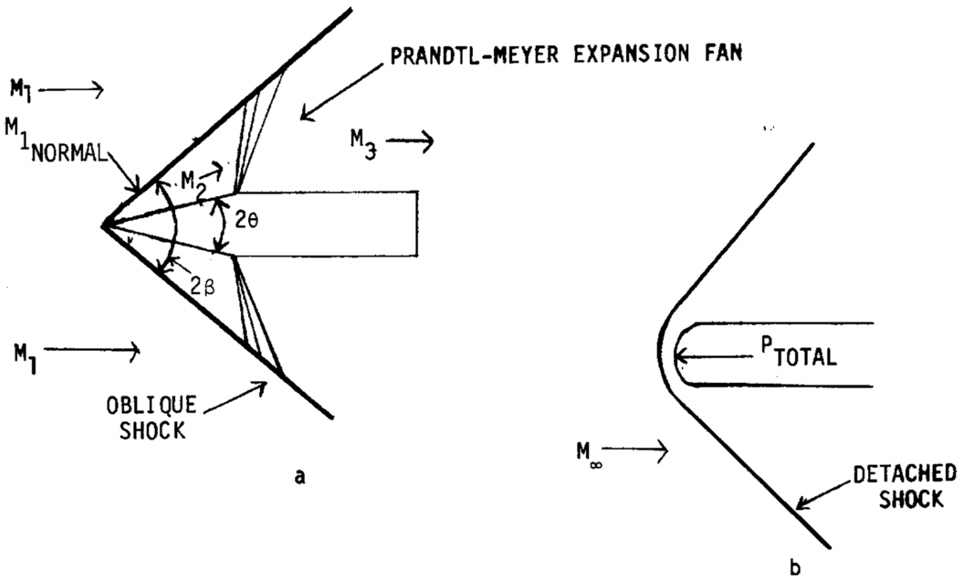


Figure 4.2: Shock Wave Structure

These equations are given graphically in many textbooks.

The relations for the conditions (pressure, temperature, density, Mach number, etc.) before and after the oblique shock wave can be determined from the normal shock relations with the modification that the incoming Mach number be replaced by the normal component of the Mach number (see Figure 4.2a). Thus,

$$M_{1_{NORMAL}} = \frac{u_1}{a_1} = M_1 \sin\beta \tag{4.7}$$

and therefore:

$$M_{2_{NORMAL}} = M_2 \sin(\beta - \theta) \tag{4.8}$$

When a blunt body is placed in a supersonic flow a detached shock wave forms in front of the body (Figure 4.2b).

Example Problem: Calculating shock angle, pressure, temperature and mach number for compression across an oblique shock wave.

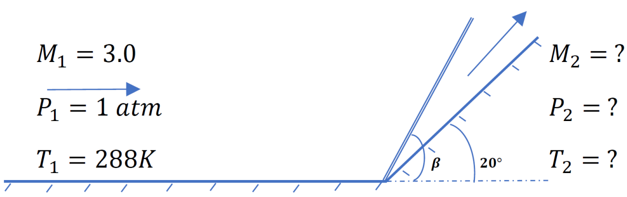


Figure 4.3: Problem Schematic for supersonic compression by turning

Solution

1. Find shock angle β

   For $M_1 = 3.0$ and $\theta = 20^\circ$

   From chart/tables for $\theta - \beta - M$ relation

   $\implies \beta = 37.76^\circ$

2. Find M_{n1}

   $M_{n1} = M_1 sin\beta = 3.0sin(37.76^\circ) = 1.839$

3. From Normal Shock Tables for $M_{n1} = 1.839$

   $\frac{P_2}{P_1} = 3.783, \frac{T_2}{T_1} = 1.562, M_{n2} = 0.6078$

4. Find P_2, T_2, M_2

   $P_2 = 3.783P_1 \implies P_2 = 3.783$
   $T_2 = 1.562T_1 \implies T_2 = 449.9K$
   $M_2 = \frac{M_{n2}}{sin(\beta-\theta)} \implies M_2 = 1.988$

Supersonic Expansion by Turning

Consider the supersonic expansion of a flow through a convex turn such as the flow shown on the wedge of Figure 4.2a. For a sharp turn (Figure 4.4) there are a series of Mach lines emanating from a single point.

Prandtl-Meyer Function The differential relation between θ and M in an isentropic compression or expansion by turning is:

$$d\theta = \sqrt{M^2 - 1}\frac{dV}{V} \tag{4.9}$$

where:

M = Mach Number
V = Velocity
θ = Angle through which flow is turned
dθ = Change in θ (positive when expanding, negative when compressing).

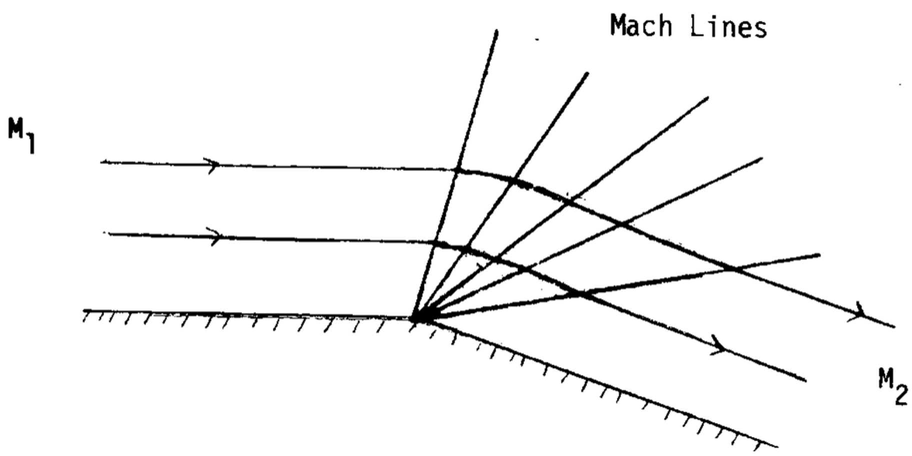


Figure 4.4: Isentropic Expansion at a Corner

Now V may be written in terms of M using the following equations:

$$V = aM \tag{4.10}$$

The relationship between a_o, a, and M is:

$$\frac{a_0^2}{a^2} = 1 + \frac{\gamma - 1}{2}M^2 \tag{4.11}$$

Using Equation 4.10 we have:

$$\frac{dV}{V} = \frac{dM}{M} + \frac{da}{a} \tag{4.12}$$

Differentiating 4.11 and using this in 4.12 we get:

$$\frac{dV}{V} = \frac{dM}{M}\left[\frac{1}{1 + \frac{\gamma-1}{2}M^2}\right] \tag{4.13}$$

Using equation 4.13 in equation 4.9 we have:

$$d\theta = \frac{\sqrt{M^2 - 1}}{1 + \frac{\gamma-1}{2}M^2}\frac{dM}{M} \tag{4.14}$$

From equation 4.14 we have:

$$\int d\theta = \int \frac{\sqrt{M^2-1}}{1+\frac{\gamma-1}{2}M^2}\frac{dM}{M} \tag{4.15}$$

Integrating Equation 4.15 we have:

$$\theta = \left(\frac{\gamma+1}{\gamma-1}\right)^{1/2}\tan^{-1}\left[\frac{\gamma-1}{\gamma+1}\left(M^2-1\right)\right]^{1/2} - \tan^{-1}\left[M^2-1\right]^{1/2} + CONSTANT \tag{4.16}$$

Define the Prandtl-Meyer function (ν) as follows:

$$\nu\left(M\right) = \left(\frac{\gamma+1}{\gamma-1}\right)^{1/2}\tan^{-1}\left[\frac{\gamma-1}{\gamma+1}\left(M^2-1\right)\right]^{1/2} - \tan^{-1}\left[M^2-1\right]^{1/2} + CONSTANT \tag{4.17}$$

Pick a value of θ and M so that the constant can be set. For $M = 1$ let $\theta = 0$ (this value of θ was chosen arbitrarily). Equation 4.16 then gives

$$CONSTANT = 0 \tag{4.18}$$

From equation 4.14 we have:

$$\int_{\theta_1}^{\theta_2} d\theta = \int_{M_1}^{M_2} \frac{\sqrt{M^2-1}}{1+\frac{\gamma-1}{2}M^2}\frac{dM}{M} \tag{4.19}$$

Thus from equation 4.19:

$$\theta_2 - \theta_1 = \nu(M_2) - \nu(M_1) \tag{4.20}$$

Let $M_1 = 1$, then $\theta_1 = 0$ (note: $\theta = 0$ was chosen to correspond to $M = 1$, i.e. $\nu(1) = 0$). Then from Equation 4.20:

$$\theta_2 = \nu(M_2) \tag{4.21}$$

Thus $\nu(M_2)$ is the angle through which the flow must be isentropically turned to obtain a Mach number of M_2 if it is initially at a Mach number of 1. The value $\theta = 0$ was chosen to correspond to $M = 1$. Any value of θ could be chosen to correspond to $M = 1$. For different values of θ a different value for the CONSTANT would be obtained. Most values of the Prandtl-Meyer function are based on choosing $\theta = 0$ for $M = 1$, i.e. $CONSTANT = 0$.

Example Problem: Calculating Mach Number after expansion wave

Suppose we have a flow with $M = M_1$ and we turn the flow through an angle $\Delta\theta$. Find the Mach number after the flow has been turned. NOTE: $\Delta\theta$ is positive in this case since the flow is expanding.

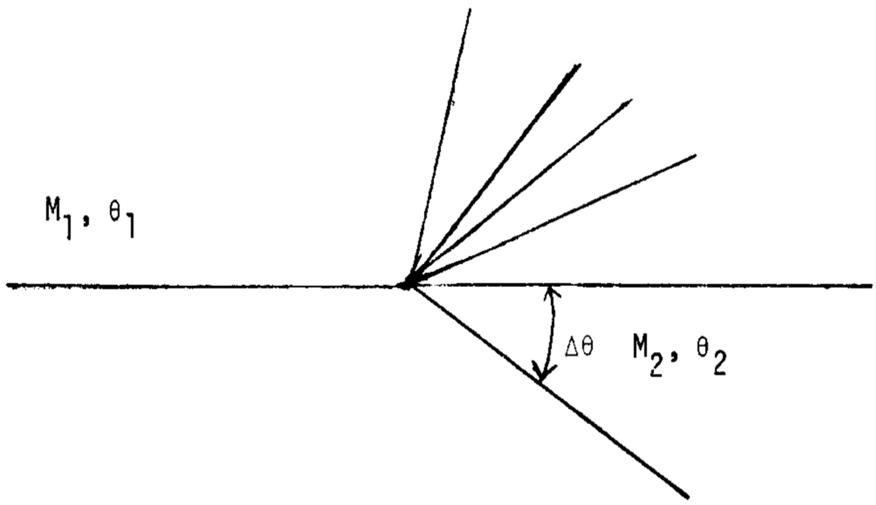


Figure 4.5: Prandtl-Meyer Expansion Fan

Solution

For $M_1 = 2.0$ and $\Delta\theta = 5$
$\theta_1 = \nu(M_1) = 26.38$ (from tables)
$\theta_2 = \theta_1 + \Delta\theta = 26.38 + 5$
$\theta_2 = 31.38$
Therefore from the tables, $M_2 = 2.185$

4.4.3 The Supersonic Wind Tunnel

A schematic of the supersonic wind tunnel used in this laboratory is shown in Figure 4.6. Compressed dry air is pumped into storage tanks with a total volume of about 350 cubic feet. These tanks hold the large amount of air that is necessary for runs of any useful duration, as the mass flow through the tunnel is quite large. The air passes through several valves and is regulated down to the pressure desired. The regulated flow then passes through a valve, which can be opened very rapidly, and into the stilling or plenum chamber, where the turbulence from the pipe flow is reduced. Because the velocity of the flow in the plenum chamber is very small, the static and total values in the plenum chamber are nearly equal. Here they will be *assumed* to be equal.

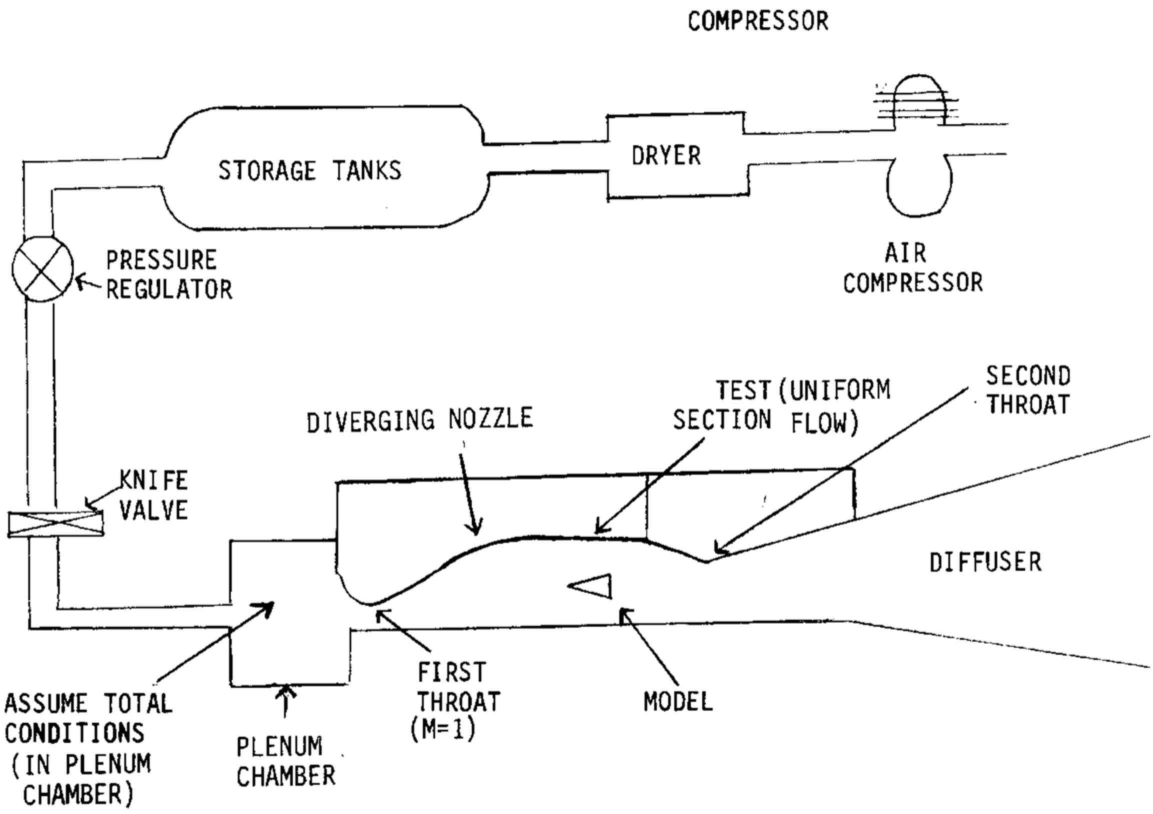


Figure 4.6: Schematic of Supersonic Wind Tunnel

The flow converges to the first throat where it reaches the sonic condition. The flow further expands and accelerates supersonically through the diverging nozzle until it reaches the test section, a region of constant area. In the test section, the flow is at a *uniform* velocity. As the air expands behind the first throat, the static temperature decreases considerably. This is why the air must be dry. If it were not, the dew point would be quickly reached and a condensation shock would form at the point where the temperature dropped below the critical value. The flow through the diverging nozzle is nearly isentropic due to the shape of the nozzle block. This shape may be computed using the method of characteristics.

As the flow accelerates supersonically down the nozzle it encounters the model. The model is located in the test section so that the air is flowing at a uniform speed as it reaches the model. A series of shock waves and expansion fans will be formed due to the disturbance of the airflow by the model. These shock waves and expansion fans are viewed using a shadowgraph or schlieren system.

Immediately behind the test section is the second throat. The flow is compressed and decelerated as the throat necks down. The center flow does not decelerate to $M = 1$, however. Behind the second throat, the flow starts to expand and accelerate. However, since total pressure has been lost because of shocks in the test section, the flow cannot accelerate for long. A normal shock forms somewhere in the diverging portion after the second throat. The flow is subsonic behind the normal shock, which is what is desired. The purpose of the second throat is to control the position of the normal shock, so that it is downstream of the test section, and as weak as possible. The diffuser downstream of the second throat reduces the losses, so that less total pressure is required to run the tunnel. In our case of a "blow down" tunnel, longer runs are then permitted.

Small pressure taps are located along the walls of the nozzle and test section. These taps allows the static pressure along the nozzle and models to be measured. A shadowgraph photographic system is used to observe and photograph shock waves in the supersonic tunnel. The shadowgraph system is an optical method for visualizing shock waves, expansion fans, and other regions of sizeable density gradients. In the experiment, it will be used to measure the oblique shock angle, β , on a wedge.

4.4.4 The Digital Pressure Transducer System

A modern multi-channel pressure transducer system will be used to measure the static pressure at the pressure taps in the tunnel walls. The PSI 9000 Series pressure measurement box is specially modified to handle the wide pressure range found in the nozzle experiment. If for any reason tubing becomes disconnected, please get the TA to reattach the tubing. This is a trivially simple step compared to replacing the box if a high-pressure tube were to be mistakenly attached to a low-pressure transducer port. Please don't hesitate to get the TA involved.
There are 16 temperature-compensated pressure transducers in the box, and one high-quality analog-to-digital converter. The 16 channels are sampled one after another, and the output is a digital signal through an RS232 port to the PC. The TA will demonstrate the software for you.

4.4.5 Supersonic Wind Tunnel Models

This lab makes use of several models to help students visualize supersonic flow over surfaces and shock and expansion waves. In this lab, we will use four models: a two-dimensional wedge, a three-dimensional cone (with the same half angle as the wedge), a thin diamond airfoil, and a blunt body. Schematics of the models with important dimensions given in Figures 4.7 to 4.10.

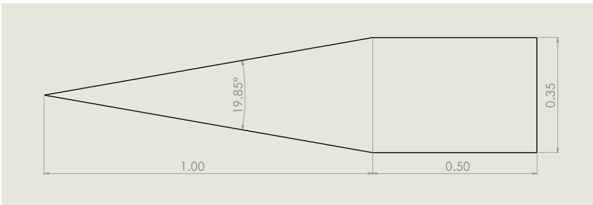


Figure 4.7: Two-dimensional wedge model.

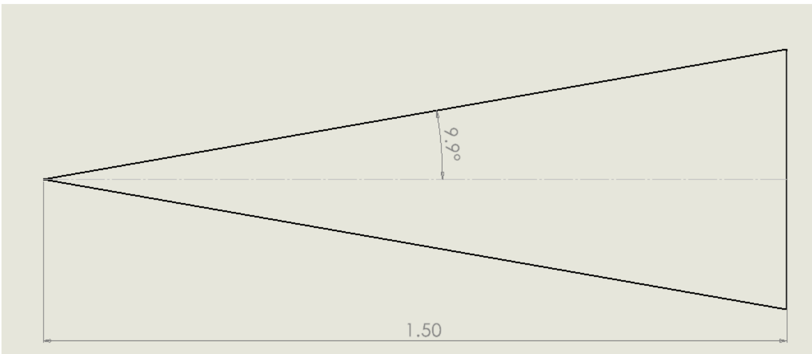


Figure 4.8: Three-dimensional cone model.

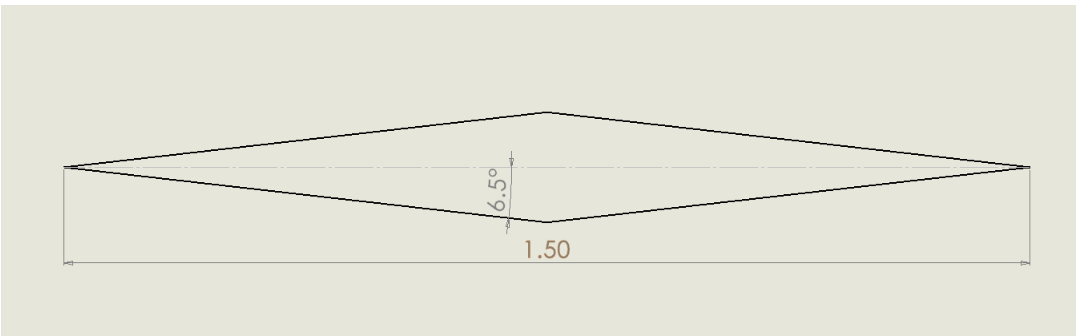


Figure 4.9: Diamond airfoil model.

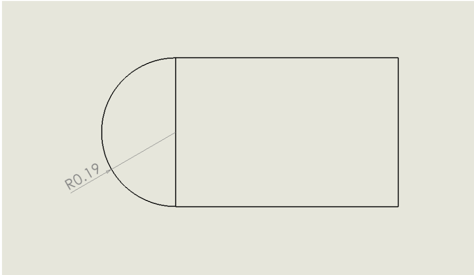


Figure 4.10: Blunt body model.

4.5 Recommended Reading

It is highly recommended that students read the following sections in Fundamentals of Aerodynamics by John D. Anderson:

- 7.6 Some Aspects of Supersonic Flow: Shock Waves
- 8.4 Special Forms of the Energy Equation
- 8.6 Calculation of Normal Shock-Wave Properties
- 9.2 Oblique Shock Relations
- 9.3 Supersonic Flow over Wedges and Cones
- 9.5 Detached Shock Wave in Front of a Blunt Body
- 10.5 Supersonic Wind Tunnels

4.6 Data To Be Acquired

1. Pressure value at which the tunnel is being operated.
2. Pressure data at various locations along the supersonic tunnel.
3. Schlieren images of flow around four different models.

Tap Number	Distance from throat to pressure tap location (in)	Height of tunnel cross-section (in)	Pressure Scanner Channel No.
1	-1.5	2.455	1
2	-0.5	0.937	2
3	0	0.796	3
4	1	0.926	4
5	2	1.1	5
6	3	1.281	6
7	4	1.447	7
8	5	1.628	8
9	7	1.883	9
10	8	1.986	10
11	9	2.074	11
12	10	2.14	12
13	11	2.189	13
14	12	2.213	14
15	13	2.213	15
16	14	2.213	16

Table 4.1: Pressure tap information. **Note that the tunnel width is 1 inch**

4.7 Lab Setup and Procedure

4.7.1 Experimental Setup and Preparation

Lab Location: ARMS B098, Supersonic Wind Tunnel "A"

Relevant Safety Hazard: The supersonic tunnel can produce high-decibel sounds when running at high speeds, which can permanently damage your hearing.

- DO NOT OPERATE THE TUNNEL WITHOUT EAR PROTECTION, even if running it at low speeds. Before running the tunnel, be sure everyone in the room has ear protection.
- Do not go past the chains into prohibited areas: this includes the area in the vicnity of tunnel exit; high speed projectiles can shoot out of the exit.
- Obey all signs and warnings posted in the lab and/or mentioned in the lab manual.

1. **Pressure Measurement and Pressure Scanner**

   1.1 ☐ Note the pressure tap locations, starting behind the nozzle throat (location of minimum area) and distributed along the diverging nozzle. The first pressure tap is near the reservoir and so the measurement will be close to the total (or stagnation) pressure.

   1.2 ☐ Note the pressure tap numbers, location relative to the throat, and corresponding pressure transducer channel number shown in Figure 4.11 and Table 4.1. Verify that the pressure channels are connected to the respective taps on the tunnel (their numbers should match). The nozzle height at each pressure tap location is also given in Table 4.1. The **tunnel width is 1 inch**, so the area (in square inches) at each tap location is simply the height (in inches) multiplied by 1.

   If for any reason the tubing becomes disconnected, ***please get the TA to reattach the tubing****. This is a trivially simple step compared to replacing the box if a high-pressure tube were to be mistakenly attached to a low-pressure transducer port. Please don't hesitate to get the TA involved.*

   1.3 ☐ Verify that the pressure scanner is turned on (check that the "PWR and "LNK" LED lights are illuminated).

   1.4 ☐ **Open the VI used for this lab.** The VI used for this lab is SWT Pressure Measurement.vi (SWT stands for "supersonic wind tunnel"). One member should log into the lab computer using their career account and navigate to the Lab 4 folder at the destination: `C:\temp\AAE33401\Lab 4 - Supersonic Wind Tunnel`. Open the LabVIEW library file `Lab4_33401_Supersonic.llb` and then open `SWT Pressure Measurement.vi`. **Please do not alter or modify any LabVIEW VIs - even very small changes can cause the program to stop working!**

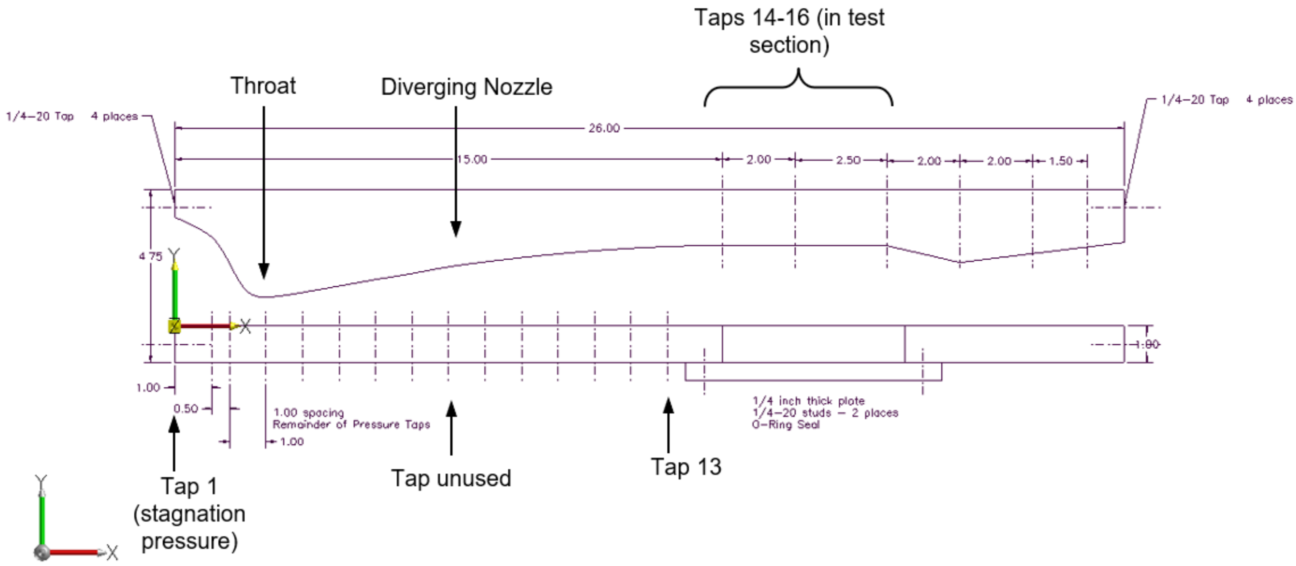


Figure 4.11: Diagram of the supersonic wind tunnel and locations of pressure taps

2. **Saving Experimental Data**

   2.1 ☐ Make a folder with your team name (e.g., Black16) in the `data_33401_lab4` folder. Make sure you save any and all data generated during the lab in this folder.

   2.2 ☐ Set the file path in the LabVIEW VI to your data folder. After the path, add a backslash and enter a name for the data file (in .txt format). For example, your final path might look like: `C:\temp\AAE33401\Lab 4 - Supersonic Wind Tunnel\data_33401_lab4\Black16\Black16_pressure_data`.

   2.3 ☐ Download and open the Lab 4 data spreadsheet from Brightspace. The spreadsheet will help you record data and perform calculations.

   2.4 ☐ Click the Run arrow ⇨ at the top toolbar to begin running the VI. The readouts from the pressure scanner channels should be fluctuating around zero. If the readouts are not fluctuating or there is an error message, get help from the TA.

   2.5 ☐ Verify that the program saves the data correctly as follows:

      2.5.1 ☐ Click the "Push to Write" button in the VI to write the pressure data to the file you specified in Step 2.2. A push is confirmed when you see the green "Success" light and when the push counter increases.

      2.5.2 ☐ Press the STOP button on the VI [STOP] to stop the program (NOT the stop sign on the top toolbar but the word STOP on the bottom left side of the main screen).

      2.5.3 ☐ Open the data file using Notepad, Wordpad or Excel. The first row of data is the pressure scanner channel number and the second row is the pressure readings from the channels at the time you clicked the "Push to Write" button. Note that these are *gauge* pressures, i.e., the difference between the static pressure at the tap and atmospheric pressure (in psi).

3. **Supersonic Wind Tunnel Operation**

   The main features of the tunnel operation are shown in Figure 4.12. The tunnel is quite loud and so it is extremely important that everyone in the room wear ear protection any time the tunnel is running. ***Before you turn on the tunnel, you must verify that everyone in the room has on ear protection.***

   3.1 ☐ Distribute ear protection to everyone in the room and close the two exit doors.

   3.2 ☐ Install the Blank Model Block in the tunnel test section. Make sure the wingnuts on the block are tightened.

3.3 ☐ Gently open the main pressurized air valve (white handle) on the supply line so that it is parallel to the pipe.

3.4 ☐ Turn on the Controller Box (red power switch). The digital pressure readout should light up and read atmospheric pressure (approximately 14.7 psi). When you are running the tunnel, the reservoir/plenum pressure (approximately equal to the total or stagnation pressure p_0) will be shown on the readout.

3.5 ☐ Set control to Remote (switch on the Controller Box). In this mode, the reservoir pressure is increased or decreased by turning the knob on the remote control. It is important to always start running the tunnel at a low speed (low reservoir pressure) to confirm that everything is operating correctly.

3.6 ☐ Reduce the pressure in the reservoir by turning the knob on the remote control counterclockwise until it stops.

3.7 ☐ Designate one team member to monitor the air supply pressure by watching the white pressure gage located on the front wall of the room next to the whiteboard.

The compressor on the building roof supplies air at $\sim$ 120 psi. As we run the tunnel at high pressures, we will deplete the supply air and the pressure will drop. **We share the air supply with the rest of the building so we should not let the pressure fall below 90 psi.** When the pressure falls to 90 psi, stop running the tunnel and wait for the supply pressure to increase.

3.8 ☐ Make sure everyone has ear protection and is wearing it correctly. Designate one team member to call out a warning that the tunnel is going to be turned on.

3.9 ☐ Turn the knob clockwise approximately two times and press and hold the red button on the remote control to run the tunnel;. **Note that the pressure readout takes time to update, so make pressure changes slowly.** Try to limit runs to 10 seconds or less to ensure the pressure tank is not depleted too much.

3.10 ☐ Release the red button to decrease the reservoir pressure back to atmospheric and stop the air flow through the tunnel. Note that the next time you press the red button, the reservoir pressure will return to the level of the last run.

3.11 ☐ When you are done operating the tunnel, turn the knob counterclockwise several times until it stops and press the red button to confirm that there is no flow.

3.12 **Schlieren Visualization and Camera Software**

3.12.1 ☐ Verify that the schlieren system light source (LED) and the camera are powered on (blinking green light on the back). If either is not turned on, notify your TA.

3.12.2 ☐ Start the camera software (PCO Camware) using the shortcut in the Lab 4 folder.

3.12.3 ☐ Under the Acquisition menu, the Live Preview mode should be grayed out; if not, click on it to enter Live Preview mode.

3.12.4 ☐ On the right-hand side of the Camware interface, under Camera Properties, set the exposure time to 5 μs.

3.12.5 ☐ Right click on the image window and select "Auto Range Peak" and "Hide Teaser".

Note that an optics setup is extremely sensitive, slight change in lens/mirror orientation can distort the image. Please do not touch the optics table. Focal length of the concave mirror used for reflecting light beam is 0.5 m ($\tilde{1}$9.6 in).

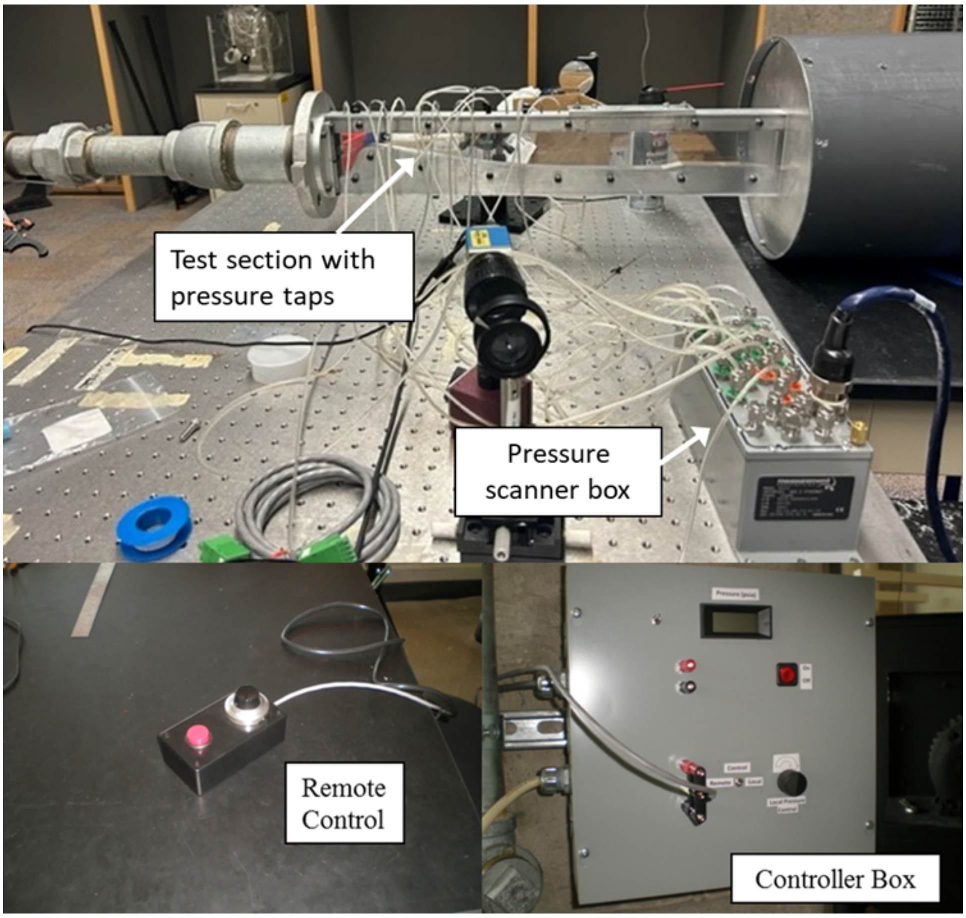


Figure 4.12: Supersonic wind tunnel and features for operation

4.7.2 Experimental Procedure

Yellow bold highlighted text indicates that student should show the data/plot obtained to the TA present at the lab.

Once the schlieren system and camera are set up and the VI is running properly, you are ready for an experiment. One team member should monitor the supply pressure, one team member should monitor the wind tunnel, model, and tubing connections, a third member can run the tunnel, and the remaining members should record the pressure data and the camera images. **Be sure everyone in the room has ear protection.**

1. **Wind Tunnel Characterization**

   You will first run experiments with no model in the test section to characterize the pressure distribution along the wind tunnel and calculate the Mach number.

   1.1 ☐ Click the Run arrow at the top toolbar to begin running the VI. You should see the channel readings fluctuating around zero.

   1.2 ☐ Slowly increase the plenum pressure to approximately 16 psi. Verify that the pressure channels are responding as you expect. Click "Push to Write" to record the pressure distribution along the tunnel.

   1.3 ☐ Slowly increase the plenum pressure to approximately 30 psi and click "Push to Write" to record the pressure data. Turn off the tunnel.

   1.4 ☐ Verify that the supply pressure has recovered to ∼120 psi, then slowly increase the plenum pressure to approximately 50 psi. Click "Push to Write" to record the pressure data. Now slowly increase the pressure to 60 psi and record the data. Turn off the tunnel.

   1.5 ☐ Copy your pressure data into the Lab 4 data spreadsheet in the appropriate worksheet. Convert the values from gauge pressure to absolute pressure.

1.6 ☐ Plot the pressure ratio p/p_0 vs. pressure tap number for the four cases ($p_0 = 16, 30, 50,$ and 60 psia). **Show your plots to your TA and discuss. Do the trends look like you expect?**

1.7 For the two latter cases ($p_0 = 50$ and 60 psia) calculate the Mach number in the test section using two methods:

1.7.1 ☐ **Theoretical Mach Number:** Calculate the predicted Mach number using the nozzle geometry, i.e., the ratio between the test section area to the throat area (A/A^*). Record your calculation in the data spreadsheet.

1.7.2 ☐ **Experimental Mach Number:** Calculate the Mach number using the experimentally measured static pressure in the test section. Use the isentropic flow equation for p/p_0 vs. M where p is the static pressure in the test section (**use Tap 15**). Calculate two values of M for the two values of p_0 and record in the data spreadsheet.

1.7.3 ☐ **Show your values to your TA. How do the experimental Mach numbers compare with the theoretical value?**

2. **Supersonic Flow Over Various Bodies**

2.1 ☐ Remove the Blank Model Block and install the wedge model (Test Article A) in the test section. Make sure to tighten the wing nuts.

2.2 ☐ You should now see the model in the field of view in the camera software. If the focus or field of view is bad, ask the TA to adjust the schlieren setup.

2.3 ☐ Reduce the pressure in the reservoir by turning the knob on the remote control counterclockwise several times. You do not need to reduce the pressure completely to atmospheric (i.e., turning the knob until it stops), but we want to start at a somewhat lower pressure and ramp up for the first model to ensure everything is operating properly.

2.4 ☐ Change the name of the pressure data file in the VI to reflect your team name and the current test model. For example, for the wedge your file name might be "Black16_wedge.txt".

2.5 ☐ Start the VI and verify that the channel readings are fluctuating around zero.

2.6 ☐ Press the red button on the remote control to start flow through the tunnel and verify that everything is operating correctly. One team member should be monitoring the pressure tap tubing and the state of the model. **If the model begins to shake rapidly or shows any signs of damage/failure, immediately turn off the tunnel and inform the TA.**

2.7 ☐ Slowly increase the plenum pressure to approximately 50 psi. You should see clear oblique shock waves emanating from the tip of the wedge and possibly expansion waves on the wedge "shoulders".

2.8 ☐ Click the "Push to Write" button to record the pressure data and save a couple of schlieren images from the camera. To capture the image, we recommend following techniques:

- Right click on the image in the PCO Camware software and select "Copy to clipboard" from the drop down menu. Paste it in a word document for further perusal.
- You may also take screenshot using the snipping tool (shortcut in the Lab 4 folder). Save the images to your data folder.

2.9 ☐ Transfer the pressure data to the appropriate tab in the data spreadsheet.

2.10 ☐ Repeat Steps 2.1-2.9 for the remaining three models: cone (Test Article B), diamond airfoil (Test Article C), and blunt body (Test Article D).You do not need to bring the pressure down to atmospheric, when testing different articles, keep the knob set at 50 psi and press the red button whenever you are ready to test the article.

2.11 ☐ After the experiment is done, turn the lights source off to prevent it from overheating.

3. ☐ **Return the experiment and the lab to its state when you first arrived.** Return all tools and equipment back to their designated location, turn off any equipment that was started for the purpose of your lab, (you may leave the PC on, and sign out of your account instead) and clean up your workspace. **Obtain approval from the TA before leaving the lab.** Thank you!

Chapter 5

Compressible Nozzle Flow

5.1 Introduction

Converging-diverging nozzles, also known as de Laval nozzles, are specialized nozzle designs commonly used in applications involving supersonic flow, such as rocket and jet engines and steam turbines. The nozzle has a distinct shape characterized by a converging section, a throat, and a diverging section. In the converging section, the cross-sectional area decreases, causing the fluid's velocity to increase as it approaches the throat, where it reaches the speed of sound (Mach 1). As the fluid passes through the throat into the diverging section, the cross-sectional area increases, allowing the velocity of the fluid to accelerate to supersonic speeds. This acceleration is accompanied by a decrease in pressure and temperature in the diverging region. The unique shape of the converging-diverging nozzle is essential for efficiently converting thermal energy into kinetic energy, maximizing thrust and propulsion efficiency in high-speed fluid flow applications.

5.2 Lab Overview

In this lab, we will visualize and characterize the flow along a CD nozzle by passing pressurized air at various pressure values.

The purpose of this experiment is to examine the flow through an isentropic, converging-diverging nozzle. The flow conditions through the nozzle will be varied by changing the total pressure in the nozzle.

The nozzle has a fixed geometry and exit area. The different flow conditions to be observed are:

1. subsonic along entire nozzle
2. normal shock-wave formation in the diverging section of the nozzle
3. normal shock at the exit plane
4. overexpanded
5. supersonic design condition
6. underexpanded

The desired flow conditions will be set by using the pressure regulator to vary the total pressure in the plenum chamber. The nozzle has a fixed geometry so the exit area is fixed. The conditions within the nozzle and at the exit will be viewed using a shadowgraph system. The static pressure variation along the nozzle will be measured using pressure taps and a pressure scanner. It is assumed that the velocity in the plenum chamber is so small that conditions in the plenum are equal to that total/stagnation conditions (i.e., $p_{plenum} = p_0$). The mass flow is calculated from the nozzle pressures.

5.3 Objectives

In this experiment you will:

1. Observe flow from a converging-diverging nozzle (Laval nozzle) using a shadowgraph for various upstream total pressures.

2. Measure the pressure distribution in the nozzle using pressure taps and a pressure scanner.
3. Measure the thrust of the nozzle as a function of total pressure

5.4 Background

5.4.1 Classification of Compressible Nozzle Flows

The various flow conditions for a converging-diverging nozzle are shown in Figure 5.1. The upper part is a schematic of the jet. The two graphs present the pressure ratio and Mach number along the axis of the jet, for various operating conditions, numbered 1 to 7 as listed below. The numerical values are given for air, which can normally be treated as a perfect gas with $\gamma = 1.4$

1. Initially as the pressure ratio, P_{amb}/P_o, is decreased from 1, the flow throughout the jet is subsonic and the exit pressure, P_{exit}, will equal P_{amb}. This exit condition is shown in Figure 5.1 (Curve 1).
2. By further decreasing the pressure ratio, P_{amb}/P_o, a point will be reached where $P_{throat}/P_o = P^*/P_o = 0.528$. At this condition the flow is subsonic everywhere except at the throat where $M = 1$. Again $P_{exit} = P_{amb}$ and the exiting flow is parallel. See Figure 5.1 (Curve 2).
3. As P_{amb}/P_o is decreased further a normal shock propagates from the throat to a point midway down the nozzle. $P_{exit} = P_{amb}$ and the flow is parallel. See Figure 5.1 (Curve 3) and flow pattern in Figure 5.2.
4. Same as Case #3 except that shock is now in exit plane. See Figure 5.1 (Curve 4) and flow pattern in Figure 5.2.
5. Decreasing the value of P_{amb}/P_o, further will expel the shock from the jet. In this case, $P_{exit} < P_{amb}$ and the flow is called **overexpanded**. The term overexpanded is used because the air is expanded beyond the point (is at lower pressure) where the pressure is matched with the exit pressure. The flow will exit the jet in a converging stream. Shocks in the exit stream will compress the flow until the static pressure of the flow equals P_{amb} (Figure 5.2). This condition is sketched in Figure 5.1 (Curve 5) .
6. When P_{amb}/P_o equals the value of P/Po calculated from the area ratio A^*/A_{exit}, the jet is said to be running at the **supersonic design (matched)** condition. Once again $P_{exit} = P_{amb}$ and the exit flow is parallel (Figure 5.2). See Figure 5.1 (Curve 6). Note: The supersonic design condition is only achieved when $P_{exit} = P_{amb}$ AND the flow is supersonic throughout the nozzle.
7. If P_{amb}/P_o is decreased further, $P_{exit} > P_{amb}$ and the flow is said to be **underexpanded**. The term *underexpanded* is used because the jet has not been expanded far enough to drop the exit pressure all the way to ambient pressure. The jet will exit in a diverging stream. Expansion fans in the exit stream will expand the flow until the static pressure equals P_{amb} (Figure 5.2). This condition is sketched in Figure 5.1 (Curve 7).

Note that the isentropic flow relations must be supplemented to correctly analyze some of these flows: the normal shock relations are needed for conditions 3 and 4, the oblique shock relations for condition 5 and the Prandtl-Meyer equation for condition 7.

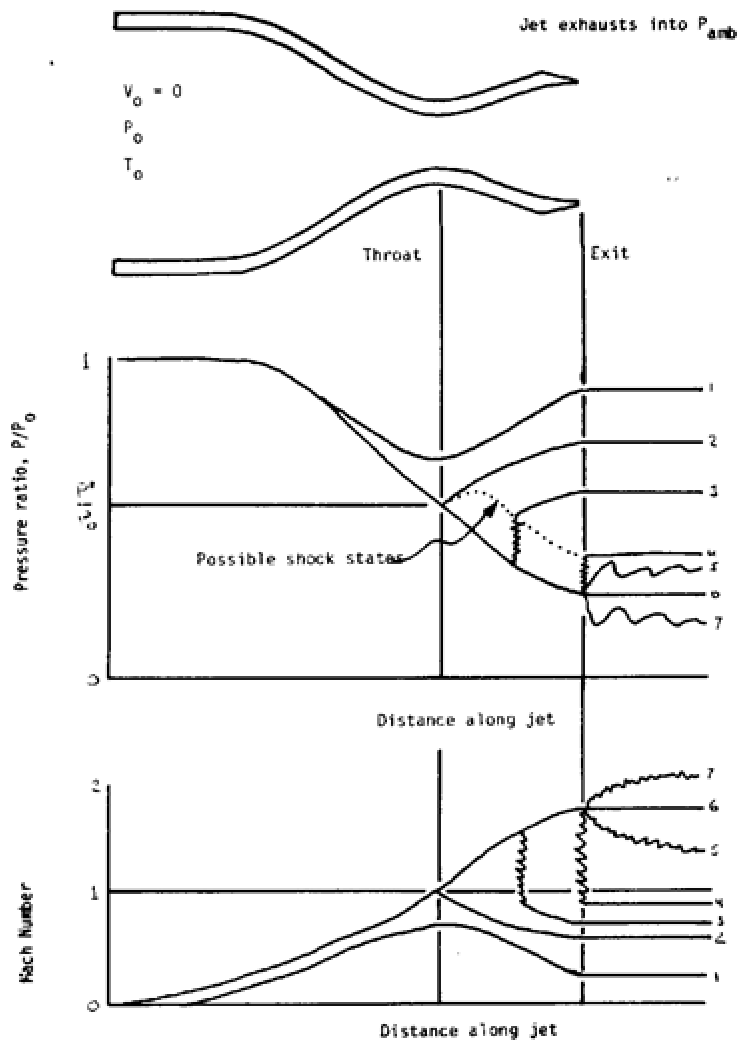


Figure 5.1: Compressible Nozzle Pressure and Mach Number Distributions

5.4.2 Jet Shock Diamonds

In the overexpanded (Figure 5.3) and underexpanded (Figure 5.4) nozzle flows, each initial oblique shock or expansion fan impinges on the opposite edge of the jet, turning the flow away from or towards the centerline. The shock or expansion fan reflects off the edge and propagates back to the other side, repeating the cycle until the jet dissipates through mixing. These flow patterns are known as shock diamonds, which are often visible in the exhaust of rocket or jet engines.

Some examples of nozzle flows are shown in Figures 5.5 and 5.6.

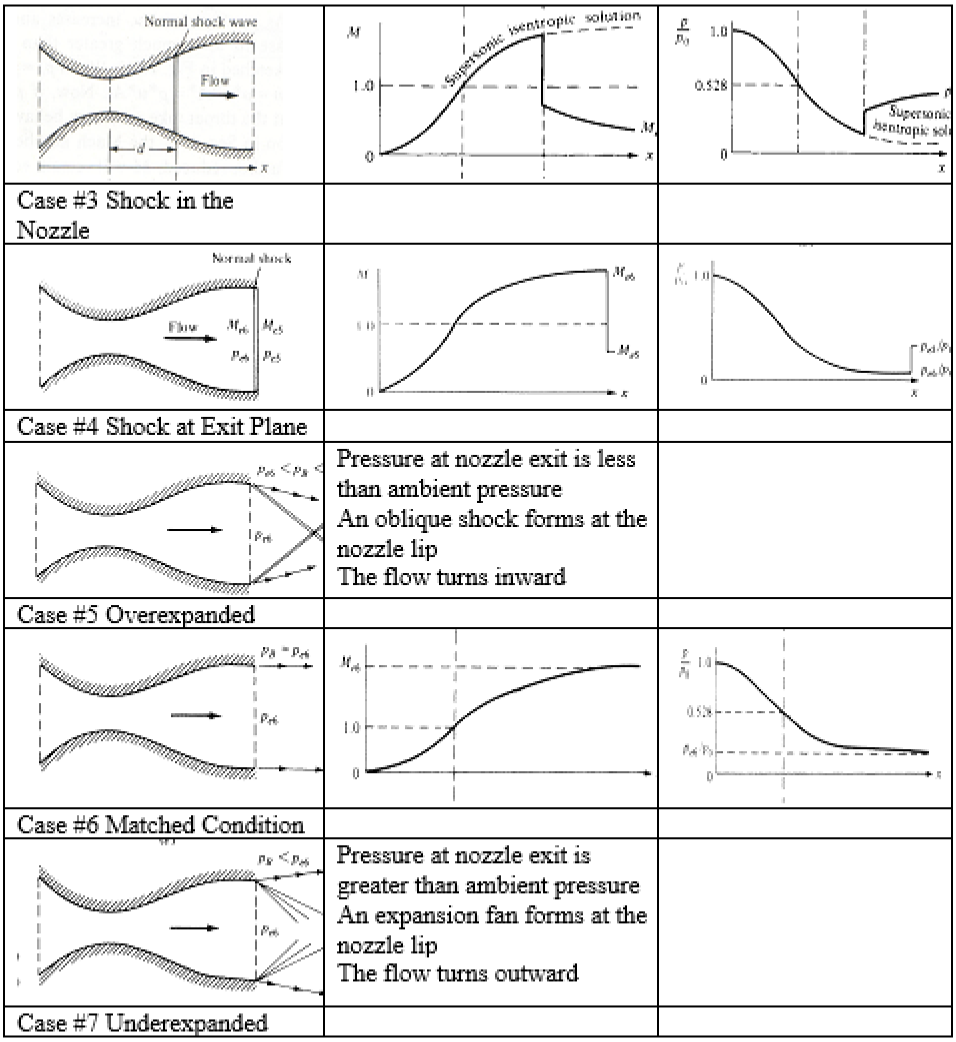


Figure 5.2: Flow Patterns in Compressible Nozzle flow. Courtesy: Fundamentals of Aeronautics, Anderson

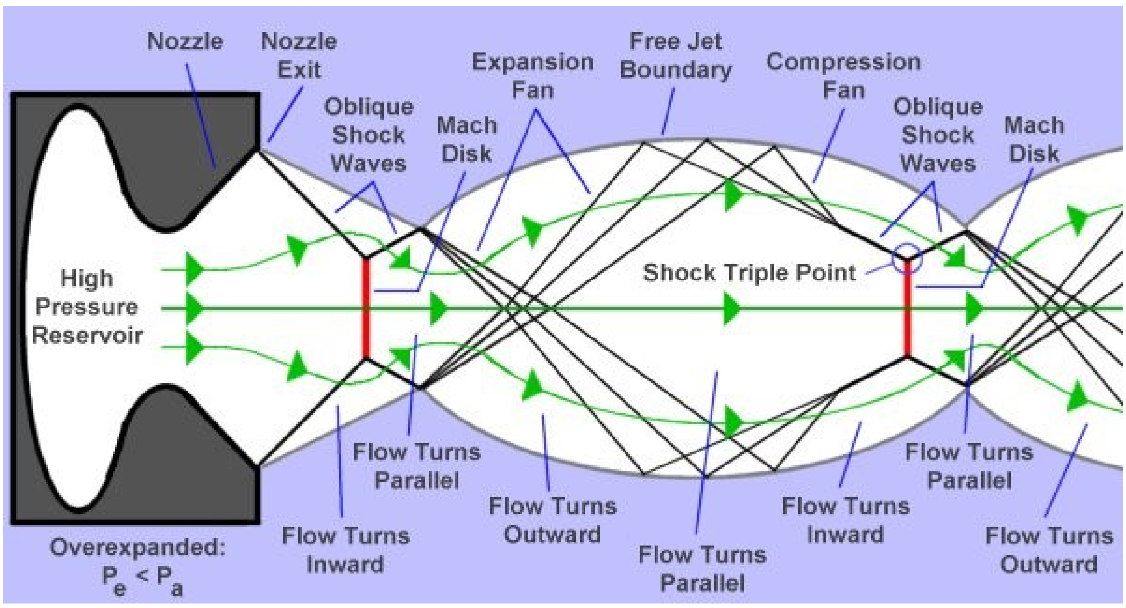


Figure 5.3: Overexpanded Nozzle Flow. Courtesy: aerospaceweb.org

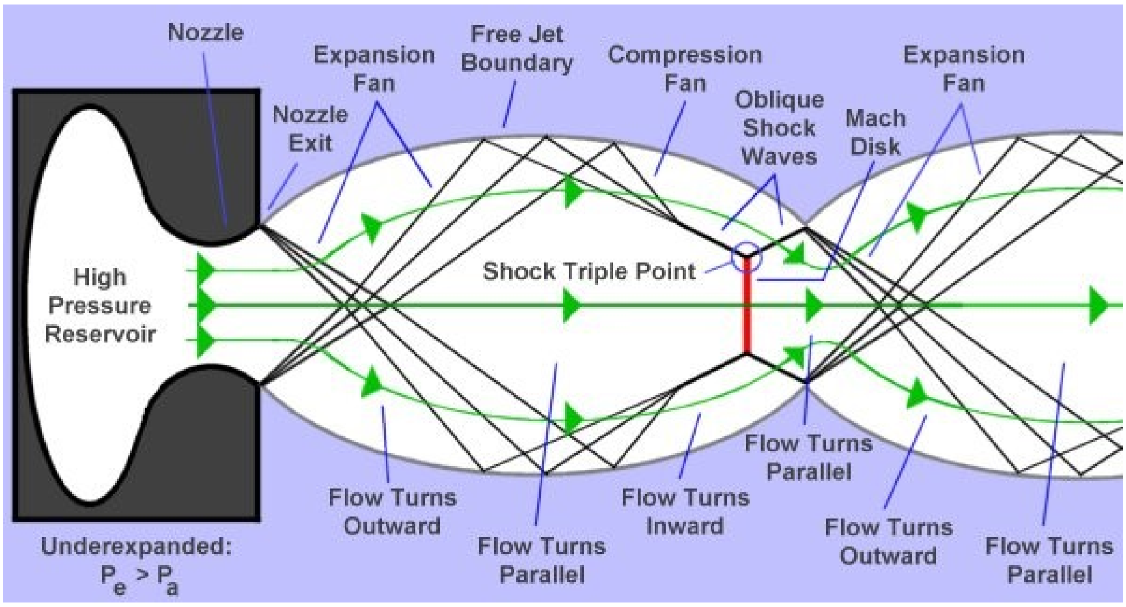


Figure 5.4: Underexpanded Nozzle Flow. Courtesy: aerospaceweb.org

Figure 5.5: Overexpanded nozzle in an SR71 as it takes off

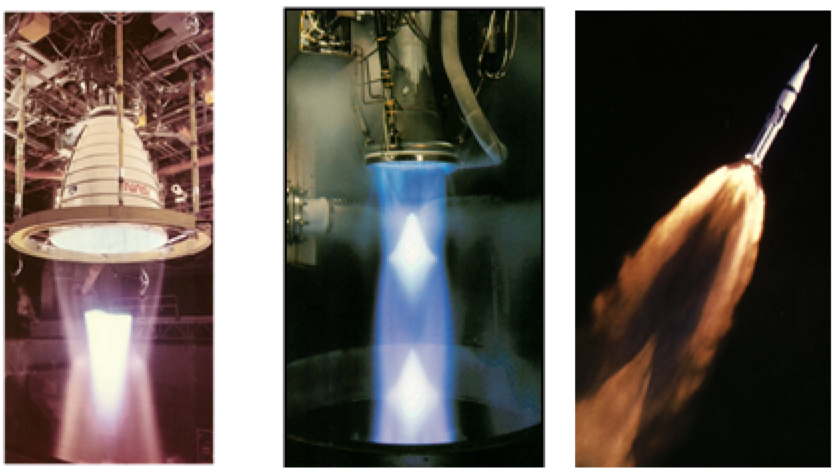


Figure 5.6: Examples of Nozzle Flow in various engines

5.4.3 Thrust

The thrust generated by the nozzle in Figure 5.7 is given by equation 5.1.

$$F = \dot{m}v_2 + (p_2 - p_3)A_2 \tag{5.1}$$

and the thrust coefficient, C_F, is then defined as in equation 5.2.

$$C_F = \frac{F}{A_t p_1} \tag{5.2}$$

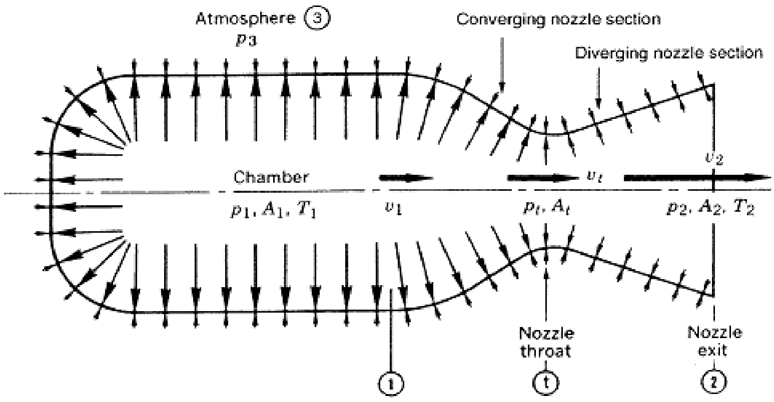


Figure 5.7: Forces on a nozzle

Note: Throughout this document,

5.4.4 Mass Flow

The theoretical mass flow, $\dot{m}$ through a nozzle can be calculated as follows:

$$\dot{m} = \rho V A = \rho^* V^* A^* \tag{5.3}$$

but,

$$\rho^* = \frac{p^*}{RT^*} \tag{5.4}$$

$$p^* = p^* \left(\frac{p_0}{p_0}\right) = \left(\frac{p^*}{p_0}\right) p_0 \tag{5.5}$$

$$T^* = T^* \left(\frac{T_0}{T_0}\right) = \left(\frac{T^*}{T_0}\right) T_0 \tag{5.6}$$

$$V^* = a^* \tag{5.7}$$

$$a^* = a^* \left(\frac{a_0}{a_0}\right)^2 = \frac{a^*}{a_0}\left(\frac{a_0^2}{a_0}\right) \tag{5.8}$$

$$a_0^2 = \gamma \cdot RT_0 \tag{5.9}$$

Therefore,

$$\dot{m} = \left(\tfrac{p^*}{p_0}\right)\left(\tfrac{T_0}{T^*}\right)\left(\tfrac{a^*}{a_0}\right)\tfrac{\gamma \cdot p_0}{a_0} A^*$$

However, the first three (3) ratios are constants,

$$\tfrac{p^*}{p_0} = 0.5283$$

$$\frac{T^*}{T_0} = 0.8333$$

$$\frac{a^*}{a_0} = \left(\frac{T^*}{T_0}\right)^{1/2} = 0.9129$$

and $\gamma = 1.4$, so

$$\dot{m} = 0.8103 \frac{p_0}{a_0} A^*$$

5.5 Recommended Reading

It is highly recommended that students read the following sections in Fundamentals of Aerodynamics by John D. Anderson:

- 10.1 Introduction (to Compressible Flow through Nozzles, Diffusers, and Wind Tunnels)
- 10.2 Governing Equations for Quasi-One-Dimensional Flow
- 10.3 Nozzle Flows

5.6 Data to be Acquired

1. Pressure at which the tunnel is being operated (plenum pressure)
2. Pressure along the tunnel cross section
3. Thrust from load cell under various test conditions

5.7 Lab Setup and Procedure

5.7.1 Experimental Setup and Preparation

Lab Location: ARMS B098, Supersonic Wind Tunnel "B"

Relevant Safety Hazard: The supersonic tunnel can produce high-decibel sounds when running at high speeds, which can permanently damage your hearing.

- DO NOT OPERATE THE TUNNEL WITHOUT EAR PROTECTION, even if running it at low speeds. Before running the tunnel, be sure everyone in the room has ear protection.
- Do not go past the chains into prohibited areas: this includes the area in the vicnity of tunnel exit; high speed projectiles can shoot out of the exit.
- Obey all signs and warnings posted in the lab and/or mentioned in the lab manual.

This experiment is conducted in Supersonic Tunnel B, which is operated differently from Tunnel A. Read more about it in Section 7.4. Tunnel B's system control and data acquisition are implemented completely in LabVIEW. The following steps will walk you through the steps to operate the tunnel and record data.

1. **Pressure Measurement and Pressure Scanner**

   1.1 ☐ Verify that the pressure scanner is turned on (check that the "PWR and "LNK" LED lights are illuminated).

   1.2 ☐ **Open the VI used for this lab.** The VI used for this lab is SWT Pressure Measurement B.vi (SWT stands for "supersonic wind tunnel"). One member should log into the lab computer using their career account and navigate to the Lab 5 folder at the destination: `C:\temp\AAE33401\Lab 5 - Compressible Nozzle Flows`. Open the LabVIEW library file `Lab5_33401_Compressible.llb` and then open `SWT Pressure Measurement B.vi`. **Please do not alter or modify any LabVIEW VIs - even very small changes can cause the program to stop working!**

   1.3 ☐ Locate the Supply Pressure readout on the LabVIEW VI. This displays the real-time output of a pressure transducer (PT1) installed on the air supply line just upstream of the electronic valve. **We should not let this pressure drop below 90 psi**. In the event that it does drop below 90 psi, the "Pressure Drop Alert" light will start flashing on the VI.

   1.4 ☐ Locate the Nozzle Reservoir/Plenum Pressure readout on the VI. This displays the real-time output of a second pressure transducer (PT2) installed in the nozzle reservoir just downstream of the electronic value. This is assumed to be the stagnation pressure for the test, and can be varied using the electronic valve, which regulates the amount of air entering the test section. This plenum pressure value is referred to as "Tap 0", and is recorded as such in the data file.

   1.5 ☐ There are 10 pressure taps along the nozzle wall. The tap numbers, location relative to the nozzle throat, nozzle height, and corresponding pressure transduce/scanner channelare given in Table 5.1. The nozzle width is constant and equal to 1 inch. High-pressure transducers (PT3 and PT4) are used for the first two pressure taps closest to the reservoir, and the remaining 8 taps are measured using a pressure scanner similar to the one used in Lab 4. The pressure readings for the 10 taps are plotted in real time in LabVIEW and will be written to a data file during the experiment.

2. **Saving Experimental Data**

   2.1 ☐ Make a folder with your team name (e.g., Black16) in the `data_33401_lab5` folder. Make sure you save any and all data generated during the lab in this folder.

   2.2 ☐ Set the file path in the LabVIEW VI to your data folder. After the path, add a backslash and enter a name for the data file (in .txt format). For example, your final path might look like: `C:\temp\AAE33401\Lab 5 - Compressible Nozzle Flows\data_33401_lab5\Black16\Black16_pressure_data`.

   2.3 ☐ Download and open the Lab 5 data spreadsheet from Brightspace. The spreadsheet will help you record data and perform calculations.

Tap Number	Pressure Transducer / Scanner Channel	Distance from throat (in)	Height of Nozzle Cross-Section (in)
0	PT2 (Plenum Pressure)	N/A	N/A
1	PT3	-1.75	0.725
2	PT4	0	0.46
3	1	0.15	0.49
4	2	0.375	0.5
5	3	0.65	0.55
6	4	0.9	0.56
7	5	1.12	0.58
8	6	1.42	0.59
9	7	1.68	0.60
10	8	1.92	0.60

Table 5.1: Nozzle pressure tap information. **Note that the tunnel width is 1 inch.**

2.4 ☐ Click the Run arrow at the top toolbar to begin running the VI. The readouts from the pressure scanner channels should be fluctuating around zero (around one for pressure transducers). If the readouts are not fluctuating or there is an error message, get help from the TA.

2.5 ☐ Verify that the program saves the data correctly as follows:

2.5.1 ☐ Click the "Push to Write" button in the VI to write the pressure data to the file you specified in Step 2. A push is confirmed when you see the green "Success" light and when the push counter increases.

2.5.2 ☐ Press the STOP button on the VI to stop the program.

2.5.3 ☐ Open the data file using Notepad, Wordpad or Excel. The first row of data is the pressure scanner channel number and the second row is the pressure readings from the channels at the time you clicked the "Push to Write" button. Note that these are *gauge* pressures, i.e., the difference between the static pressure at the tap and atmospheric pressure (in psi).

3. **Supersonic Wind Tunnel Operation**

The tunnel is quite loud and so it is extremely important that everyone in the room wear ear protection any time the tunnel is running. ***Before you turn on the tunnel, you must verify that everyone in the room has on ear protection.***

3.1 ☐ Distribute ear protection to everyone in the room and close the two exit doors.

3.2 ☐ Gently open the main pressurized air valve (white handle) on the supply line so that it is parallel to the pipe.

3.3 ☐ Turn on the tunnel using the "Tunnel Switch" button. The Tunnel ON light should turn green. The reservoir pressure should read upwards of 120 psig and the plenum pressure should show about 14.7 psia (or atmospheric pressure). When you are running the tunnel, the plenum pressure (approximately equal to the total or stagnation pressure p_0) will be shown on the readout.

3.4 ☐ It is important to always start running the tunnel at a low speed (low plenum pressure) to confirm that everything is operating correctly.

3.5 ☐ Supply pressure should not drop below 90 psi. The compressor on the building roof supplies air at $\sim$ 120 psi. As we run the tunnel at high pressures, we will deplete the supply air and the pressure will drop. **We share the air supply with the rest of the building so we should not let the pressure fall below 90 psi.** When the pressure falls to 90 psi, stop running the tunnel and wait for the supply pressure to increase.

3.6 ☐ Make sure everyone has ear protection and is wearing it correctly. Designate one team member to call out a warning that the tunnel is going to be turned on.

3.7 ☐ Increase voltage of the reservoir pressure valve control slowly to around 2V and visualize the pressure variations on the LabVIEW VI. Try to limit runs to 10 seconds or less to ensure the pressure tank is not depleted too much. Use the slider for coarse adjustment and the precise numeric control arrows for finer adjustment of valve voltage.

3.8 ☐ Bring the slider back to 0 V

3.9 ☐ When you are done operating the tunnel, turn the valve control to 0 V and press the tunnel switch to turn the tunnel off. Then stop the program.

4. **Schlieren Visualization and Camera Software**

4.1 ☐ Verify that the schlieren system light source (LED) and the camera are powered on (blinking green light on the back). If either is not turned on, notify your TA.

4.2 ☐ Start the camera software (PCO Camware) using the shortcut in the Lab 5 folder.

4.3 ☐ Under the Acquisition menu, the Live Preview mode should be grayed out; if not, click on it to enter Live Preview mode.

4.4 ☐ On the right-hand side of the Camware interface, under Camera Properties, set the exposure time to 5 μs.

4.5 ☐ Right click on the image window and select "Auto Range Peak" and "Hide Teaser".

5.7.2 Experimental Procedure

Yellow bold highlighted text indicates that student should show the data/plot obtained to the TA present at the lab.

Once the schlieren system and camera are set up and the VI is running properly, you are ready for an experiment. One team member should always be monitoring the Supply Pressure. **Be sure everyone in the room has ear protection.**

1. ☐ Click the Run arrow at the top toolbar to begin running the VI. You should see the channel readings fluctuating around zero.

2. ☐ The following flow conditions are to be found and the appropriate data taken. Identify the flow conditions by observing the schlieren visualization and pressure distribution. Reference should be made to the pressure distribution sketches shown in Figure 5.1 and the flow field sketches in Figure 5.2 in the lab background document in order to obtain the desired flow.

2.1 ☐ Subsonic flow along the entire nozzle.

2.2 ☐ Flow such that a normal shock wave has formed about midway down the nozzle.

2.3 ☐ Flow such that a normal shock wave has formed at the exit plane.

2.4 ☐ Overexpanded flow.

2.5 ☐ Perfectly expanded flow (nozzle design condition).

2.6 ☐ Underexpanded flow.

3. ☐ Run the tunnel by following the steps listed previously and increase the valve voltage slowly. When you obtain the desired flow (visualized by the schlieren image for various test cases 2.1 through 2.6), record the following:

3.1 ☐ Plenum total pressure (shown as a digital readout, manually record these in the data spreadsheet)

3.2 ☐ Static pressure distribution from the pressure taps and scanner/transducer (use "Push to Write" to record the data, transfer the data to the data spreadsheet)

3.3 ☐ At least one shadowgraph image (using the PCO Camware software)

3.4 ☐ Thrust (shown as a digital readout, manually record these in the data spreadsheet).

Make sure all values are recorded at appropriate locations in the Lab 5 data spreadsheet.

4. ☐ Measure the thrust as a function of the plenum pressure in 5 psi increments from 25 to 65 psi and record them in Lab 5 data spreadsheet. **Tare the load cell before starting to increase the pressure.** Produce a plot of thrust as a function of plenum pressure.

5. ☐ **Return the experiment and the lab to its state when you first arrived.** Return all tools and equipment back to their designated location, turn off any equipment that was started for the purpose of your lab, (you may leave the PC on, and sign out of your account instead) and clean up your workspace. **Obtain approval from the TA before leaving the lab.** Thank you!

5.8 Additional Information

5.8.1 Additional Notes on Nozzle Flow Conditions

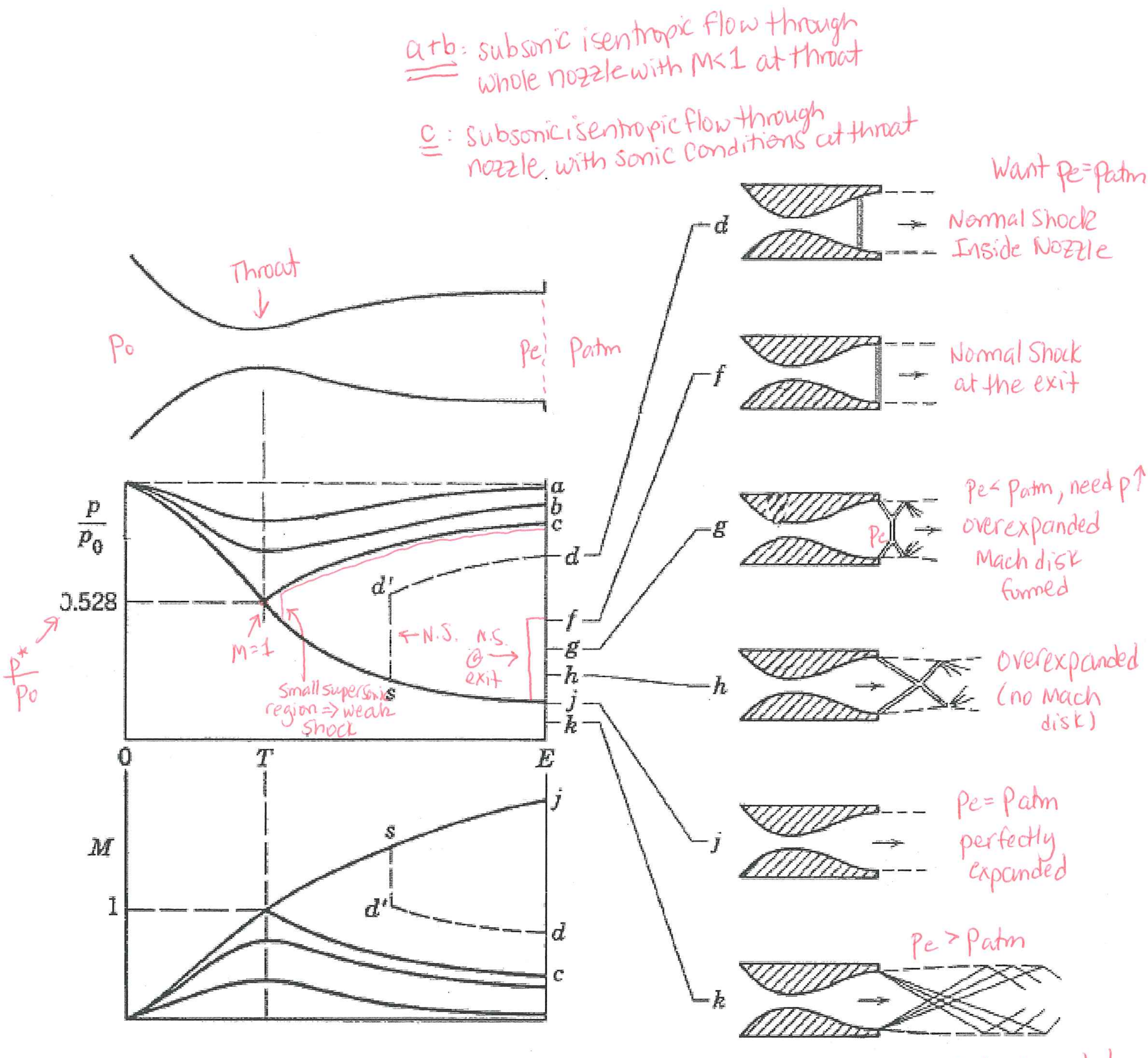


FIG. 5·3 Effect of pressure ratio on flow in a Laval nozzle.

Chapter 6

Design your Own Experiment

6.1 Introduction and Lab Logistics

Lab 6 involves several steps outlined below (chronologically). Important dates are listed on the course Brightspace Calendar and in the syllabus. Highlighted parts of the text have relevant content and/or links in Brightspace, navigate to "Content > Lab 6 - Design Your Own Experiment".

1. ☐ **Proposal Development:** As a team, develop a proposal for your Lab 6 experiment using the template provided in Brightspace ("Lab 6 Proposal Template"). Proposals that do not follow the template will not be considered for evaluation. Based on the feedback you receive, you will be required to submit a revised proposal (please see the grading rubrics in Section 6.2). Student teams are required to submit a draft proposal and a revised proposal (after receiving feedback).

   Remember that a proposal is different than a standard lab assignment - the primary goal of a proposal is to convince the reader to accept and "buy in" to your ideas. Proposals are written frequently in industry and academia, for example, to help organizations obtain funds to operate. Stakes are often high in those instances, as an accepted proposal can bring millions of dollars into the organization. Always proofread and perform appropriate grammar checks before submitting a proposal.

2. ☐ **Proposal Submission:** Submit the proposal in pdf format as a group in Gradescope. Instructions for submitting a group assignment can be found here: Group Submission on Gradescope. You must also propose **3 potential times over the weeks before the final deadline** for completing your lab when submitting this proposal. Please base your proposed times on your team members' availability and a discussion with your TA. View the current schedule of various teams/experimental rigs on Brightspace ("Lab 6 Schedule"). Refer to "Schedule Testing time"(7) for more information.

3. ☐ **Preliminary Proposal Review:** Your TA will review your proposal and provide feedback as annotations in Gradescope within 2 to 3 days.

4. ☐ **Instructor Feedback:** The course instructor will briefly meet with each team to discuss their proposal and provide additional feedback.
   - Due to the large number of teams, meetings will be limited to 10 minutes.
   - At least 2 members of the team should be available to attend.
   - Schedule a meeting time (only one member needs to complete the online form, **sign up using the link in Brightspace**).
   - The goal is that by the end of this meeting, you will have a concrete plan for completing your project.

5. ☐ **Final Proposal submission** After the first round of review, your team will be needed to resubmit your proposal after incorporating the feedback from the instructor and your TA. Upload the revised proposal on gradescope.

6. ☐ **Fabrication/acquisition of test models:** Based on your proposal, you may need a physical model that can be tested in one of our facilities. The available test section dimensions for each facility are given in the table 6.1.

Facility	Location	Test Section Dimensions		
		Length (in)	Breadth (in)	Height (in)
Low-Speed Wind Tunnel	ARMS B098	14	14	14
Water Table	ARMS B098	8	10.5	6
Supersonic Tunnel	ARMS B098	1.625	0.875	1.25
Boeing Wind Tunnel	AERO	72	72	48

Table 6.1: Test Section Dimensions of various testing facilities

There are several options for acquiring or fabricating your test models:

6.1 *Ready-made models*: Use an off-the-shelf model for your test, e.g. a toy car, an action figure, sports balls, etc. We also have 3D printed airfoils compatible with the Armstrong subsonic wind tunnel, for the following NACA configurations, some of which have commonly been used in AAE 33300 lecture course: NACA0012, NACA2412, NACA2415, NACA4412 and NACA66(2)-215 (see figure 6.2. We also have a collection of models used by teams in the past. They are situated near the supersonic tunnel area that you are free to use (see figure 6.1). If you want to test a model that is not already available, we can purchase it for you (refer to section "Purchasing")

Figure 6.1: Models from previous semesters that are available to use

(a) Various NACA airfoils with endplate

(b) Isometric view of the airfoils with mounting holes

Figure 6.2: 3D printed NACA airfoil available for Lab 6 testing

6.2 *3D printing:* You may choose to 3D print a model for testing. Purdue has several options available for 3D printing:

- 3D print at Purdue AAE (preferred): Please read more about Purdue AAE 3D printing in section 6.3.1. Contact **Adarsh Agrawal** (agraw156@purdue.edu) for assistance in using the Purdue AAE 3D printers. Filament will be provided.
- 3D print at WALC (and others): For other printing options, see: `https://guides.lib.purdue.edu/3dprinting/Home`. Note that some of these options have a weight limit per print. We cannot

provide filament for any print done outside Purdue AAE.

- 3D print on your own: You can print your 3D model outside of Purdue. Again, we cannot provide filament for any prints done outside Purdue AAE.

6.3 *Foam cutting:* If you want to test airfoil/wing models, you can make them using the CNC foam cutter in the Design, Build, Test (DBT) lab (ARMS 2098). Please contact **Tom Bietsch** (tbietsch@purdue.edu) for scheduling a foam cutting session. Read the Foam Cutting Guidelines in section 6.3.2 before scheduling. Foam material will be provided.

6.4 *Machining:* In rare circumstances, you may need to have your model machined. Note that the lead time is high for machining so this option should be considered only when absolutely necessary. Bechtel Innovation Design center offers machining facilities, and AAE's AERO (Aerospace Sciences Lab) has machining services available. Please contact **Rob Hughes** (rrhughes@purdue.edu), AND Dr. **Sally Bane** (sbane@purdue.edu) if you think your experiment requires machining.

6.5 *Something else:* At Purdue AAE, the sky is not the limit. You may think outside the box and create something entirely different which may or may not include any of the above. Reach out to your TA if you want feedback on your ideas.

- **Purchasing**

We can purchase models or materials for your team, place request using the link on Brightspace "Lab 6 Purchase Request".

Purchase requests can include ready-made models, raw material (e.g., plywood, PVC pipe, etc.), specific fasteners/hardware, etc. There is no set budget for each team, but **please keep costs below $30 if possible.**

- **Mounting Considerations**

For almost anything that is being tested in the tunnels, mounting the models and understanding clever ways to do it need to be explored. While designing your model(s), you should also plan on how to mount the model in the experimental facility. Useful information is provided in the Mounting Guidelines section. If you have questions about mounting your model or need help with simple drilling, cutting, soldering, etc., contact the AAE Lab Technician **Rob Hughes** rrhughes@purdue.edu.

- **Emailing for help**

All correspondence sent out to any staff/faculty regarding Lab 6 must have the following subject line: "Lab 6 AAE33X01 Team YZ Color". X refers to the course code (3 for AAE33301, 4 for AAE33401), "YZ" is your team number, and "Color" is your team color (Black or Gold).

All requests for help with fabricating and/or mounting models, purchases, and other appointments with staff must be submitted by a <u>specific date listed in the Brightspace calendar.</u>

7. **Schedule testing time:** Contact your lab TA to schedule time to perform your experiments in the desired facility. Slots may become limited over time, so make sure you reserve your time as early as possible.

7.1 If necessary, your TA will schedule a virtual meeting with your team to discuss and finalize details for completing your project.

7.2 You will need to find times that work for your team, your TA, and during which the equipment you need is available.

7.3 Typically, teams like to perform their experiments during their usual lab time, but do not assume that this will be possible since other teams might reserve the equipment first.

7.4 The schedule for the subsonic wind tunnel in ARMS fills up quickly, so please schedule early!

7.5 You can access the "Read-only" lab schedule for various equipment for lab 6 on Brightspace. **Only your TA can schedule your lab 6 time for you.**

7.6 Fill out the Qualtrics survey "Lab 6 Slot Requester" with 3 potential times and the equipment you intend to use.

8. **Prepare and submit your Lab 6 report:** Prepare a complete technical report as a team and submit it in Gradescope within 1 week of completing your testing in the lab. Refer to the "Guidelines on Writing Lab Report" on Brightspace. Please confirm the due date with your TA.

9. **Complete the peer evaluation:** You must complete the online peer evaluation within 2 days of submitting the final report. You will receive an email from CATME regarding this. Please login to CATME.org tool, at `https://catme.org/login/index` to get started.

6.2 Rubrics

Lab 6 will be graded on the following components:

1. Draft Proposal: 5 points
2. Final Proposal: 20 points
3. Report: 100 points
4. Peer evaluation: 40 points

Total Lab 6 score: 165 points, 25% of your final grade

The point distribution for the proposal and report can be viewed in their respective templates that are uploaded on Brightspace.

6.3 Additional Services Information

6.3.1 3D printing at Purdue AAE

Point of Contact: Adarsh Agrawal (agraw156@purdue.edu)

3D printing has become an extremely common methodology for prototyping/manufacturing quickly, efficiently, and accurately. Although you can 3D print using a variety of materials, Polylactic acid (PLA) is widely used in plastic filament in fused filament fabrication (FFF) technology of 3D printing.

We have the following 3D printers available:

1. Bambu A1 Mini: Print Volume: 180 × 180 × 180 mm.
2. Bambu P1S: Print Volume: 256 × 256 × 256 mm.
3. Bambu X1 Carbon: Print Volume: is 256 × 256 × 256 mm.

Make sure that the maximum dimension of your model is within the volume limits listed above. We will use PLA for printing in these printers. There is no limitation on filament use on these printers and it will be provided by us. If you are building your own model, we would need an STL file which will then be programmed in the printers. If you are unsure about your design and would like assistance on making design modifications, please send the original part file to the point of contact. Read about the mounting guidelines in section 6.3.3, to incorporate the mounting holes within your model (drilling into 3D printed model is not advised).

Note that 3D printing is not straightforward. Unlike the popular notions like "anything can be 3D printed" or "you can probably print a 3D printer using another 3D printer," 3D printing requires some foresight. Since the FFF technology depends on depositing layers on top of each other to create a part, one needs to make sure that the design has fewer overhangs. If there are any, proper supports shall be provided to accommodate it.
Please email your design to the point of contact to get started.

File Format: You might already have a CAD design worked up for the model you desire to 3D print. We'd like you to send the file in these formats, with decreasing order of preference: .SLDPRT, .PRT, .STEP, .STP, .STL. We generally like to get the part file and create the .STL ourselves. Avoid sending only the .STL files since these files are unitless in nature and it is hard to modify the part.

Note: If you end up printing the models yourself, make sure you provide at least two to three walls near the mounting hole if your model to make sure that we can tap the hole.

6.3.2 Foam Cutting Guidelines

Point of Contact: Tom Bietsch (tbietsch@purdue.edu)

The AAE Design-Build-Test Lab (ARMS 2098) has a CNC foam cutting machine that can be programmed to cut custom profiles. If you want to use the machine to make airfoil/wing models for testing, gather the following information before reaching out to the point of contact:

1. Airfoil profile you wish to test.
2. A .dat file containing the airfoil profile coordinates, which can be found online at www.airfoiltools.com. Save the .dat file to an external USB drive to transfer it to the foam cutter computer.
3. Have an idea about the chord length, span, and the maximum thickness of the airfoil you wish to create. Note that since the foam sheets are 2 inches thick, the maximum thickness must be 1.75 inches or less.
4. If you are planning to test your airfoil/wing in the ARMS Low-Speed Wind Tunnel, please note these additional design considerations:
    - The chord length of your model should not exceed 11.75 inches if you wish to test it in the Armstrong Wind Tunnel.
    - The span of the model can be no more than 12 inches.
    - For airfoil models, you should attach end plates to minimize the wing tip effects. You can cut these out of cardboard and glue them onto the foam.

6.3.3 Mounting Guidelines

Point of Contact: **Rob Hughes** (rrhughes@purdue.edu)

Low Speed Wind Tunnel at Armstrong

The "adjustable angle of attack" test stand has several options for mounting your model. The stand has two separate mount heads:

1. One with fixed studs (DO NOT take apart the mount head with fixed studs) which can facilitate material thickness up to 7/8" (22.3mm). This mount head is being referred to as Main head.
2. The other mount head has screw holes for material thicknesses from 1" (25.4mm) and above. This mount head is being referred to as Auxiliary head. Refer to the figure below.

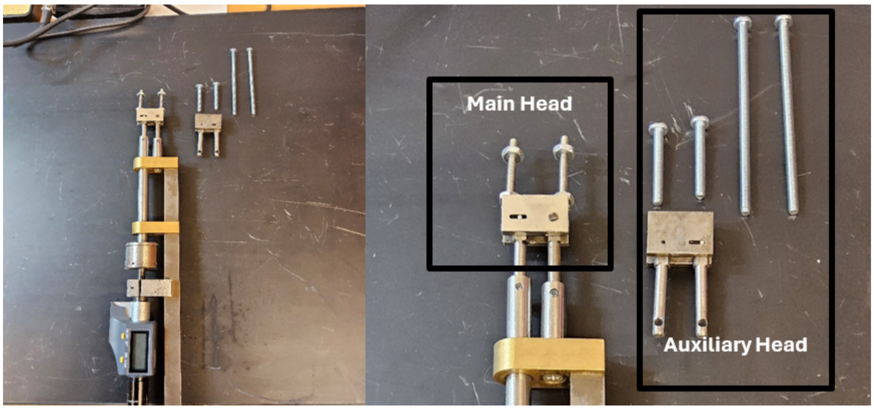


Figure 6.3: Main head and auxiliary mount head for mounting models which need change in Angle of Attack

The two heads are interchangeable on the test stand using two hex key screws located on the side of the linear shafts. If your model will not mount using the studs on the main head then remove the two hex key screws and change the head. Again, **do NOT try to remove the studs from the main head**. Refer to the figure below for more information.

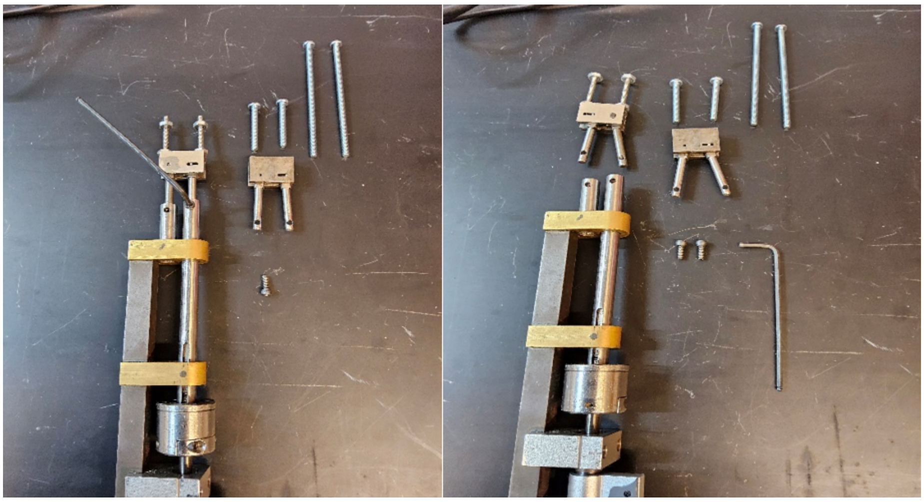


Figure 6.4: Detaching studs from airfoil stand

Both mount heads have a 0.82-inch (20.828mm) center to center distance between fasteners and are orientated parallel to the air flow of the tunnel. The Diameter of the studs on the main head is 0.142 inch (3.62mm) and a 9/64 drill bit works perfectly. The diameter of the screws for the auxiliary head is 0.171 (4.34mm) and a 11/64 drill bit works for this. Refer to the figure below.

Note for 3D printing: Please print the holes of diameter 0.145 in to take into account the contraction caused by 3D printing when using the fixed stud (0.142 in diameter stud), and print holes of diameter 0.174 in when using the detachable stud (0.171 in diameter stud).

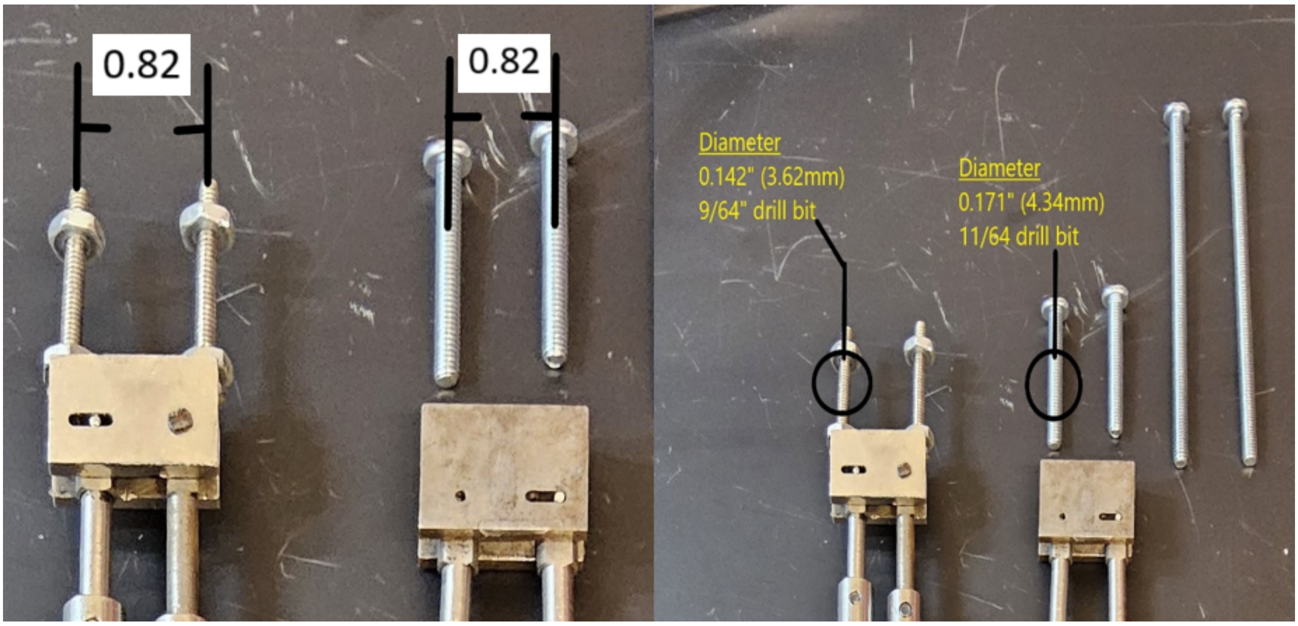


Figure 6.5: Dimensions related to mounting head

Important notes

1. Do not try to remove the studs from the main mount head.
2. Do not modify mount stands/heads in any way.
3. Do not disassemble any test/mount stands from other lab projects.

If your model requires any custom mounting considerations outside of these guidelines or if you would like any help in drilling/mounting your models contact Rob Hughes (refer to the email given at the beginning of this section for contacting him).

Supersonic Wind Tunnel

There are two model mount blocks available for use in the supersonic wind tunnel. Each has a small threaded stud to screw a model onto. One has an 8-32 threaded stud and the other has a 10-24 threaded stud. Refer to the figure below. If you want to make a model to mount on either of these mounts, include an untapped hole in your model. **For 8-32 stud, use diameter 0.137 in, for 10-24 stud use 0.15 in.**. The depth of the hole needs to be at least 0.5 in. We will use these mounts when the size of the model is around 1 in.

If the size of the model exceeds 1 in, we will have to mount it from the top.

In some cases, a mount block can be custom made to facilitate a special mounting circumstance; in this case, you must have the approval of your Professor and AAE Lab Tech. It is not advisable to put 3D printed models in the supersonic tunnels. We can help develop a machined model for testing purposes. If you want to propose a project in the supersonic tunnel, please contact Prof. Bane and Rob Hughes as early as possible to discuss feasibility.

Important notes

1. Special care should be taken not to overtighten models onto mount blocks.
2. Do not disassemble any models/blocks from other lab projects.

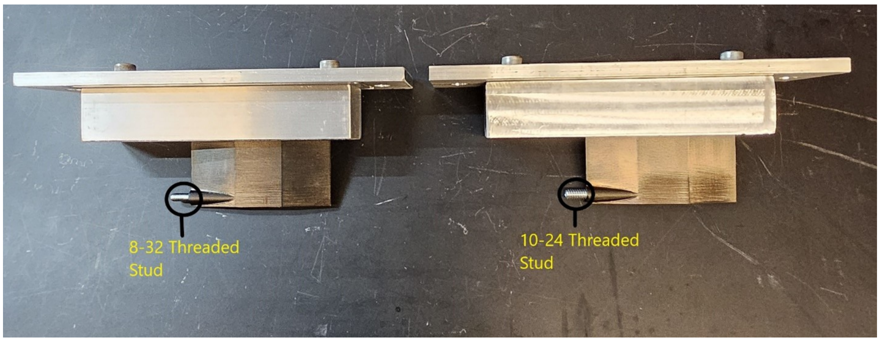


Figure 6.6: Mounting mechanisms for supersonic tunnel

3. Do not modify mount blocks or model mounts in any way.

Water Table

Currently, no specific guidelines exist for model mounting in the water table. All mountings for these locations are considered custom and will have to be approved by your Professor and the AAE Lab Tech.

6.3.4 Boeing Wind Tunnel

The following hardware is utilized to set up a generic experiment at the Boeing Wind Tunnel:

1. Shoulder Screw: #10-24 thread size, 1/4 in. shoulder length, figure 6.7:
`https://www.grainger.com/product/Shoulder-Screw-Std-Shoulder-1CA69`

Figure 6.7: #10-24 thread size, 1/4 in. shoulder length

2. Shoulder Screw: #10-24 thread size, 3/8 in. shoulder length, figure 6.8:
`https://www.grainger.com/product/Shoulder-Screw-Std-Shoulder-5YPC9`

Figure 6.8: #10-24 thread size, 3/8 in. shoulder length

3. Button Screw #10-24 thread size, 0.5 in. length, figure 6.9:
`https://www.grainger.com/product/Socket-Head-Cap-Screw-10-24-1BU82`

Figure 6.9: #10-24 thread size, 0.5 in. length

4. Hex Nut: #10-24, hex width of 3/8 in., figure 6.10:
`https://www.grainger.com/product/Hex-Nut-10-24-Thread-2GA58`

Figure 6.10: #10-24 hex width, 3/8 in. length

5. Hex Key: 1/8 in. tip size, figure 6.11:
`https://www.grainger.ca/en/product/WRENCH%2C-HEX-KEY-ALLEN-WRENCH-1-8IN/p/APOWH1/8`

Figure 6.11: Hex Key, 1/8 in. tip size

6. Magnetic Digital Inclinometer: this instrument can help find the inclination of the airfoil. It must be zeroed out before use, by using chord of the airfoil as a baseline.

Chapter 7

Appendix: Operation Manuals

For the sake of keeping lab background and procedure documents succinct and precise, additional information pertaining to various parts of the lab has been added here. You will refer to various sections of this chapter, based on what experiment you are conducting in the lab.

7.1 Armstrong Low-Speed Wind Tunnel

The Wind Tunnel in Neil Armstrong Hall of Engineering (ARMS) is located in Room B098 in the basement level of the building. The wind tunnel is designed and constructed by Dr. John P. Sullivan who is an Emeritus Professor at the School of Aeronautics and Astronautics.

The wind tunnel has two test sections, with cross-sectional dimensions of 1 ft by 1.5 ft (12 in by 18 in or 3.66 m by 5.49 m). It can run at a max speed of around 40 m/s. The tunnel has a rectangular intake of dimension 40 by 36 inches, which supplies air to a test section of size 12 by 18 inches. The wind tunnel is controlled by a variable frequency drive which drives a 240 V, 15 hp motor and has a squirrel cage type blower wheel.

This provides a contraction ratio of about 6.3. The contraction is a pair of matched conics, matched at X/L = 0.5.

The contracting intake portion and part of test section of the tunnel is shown in figure 7.1

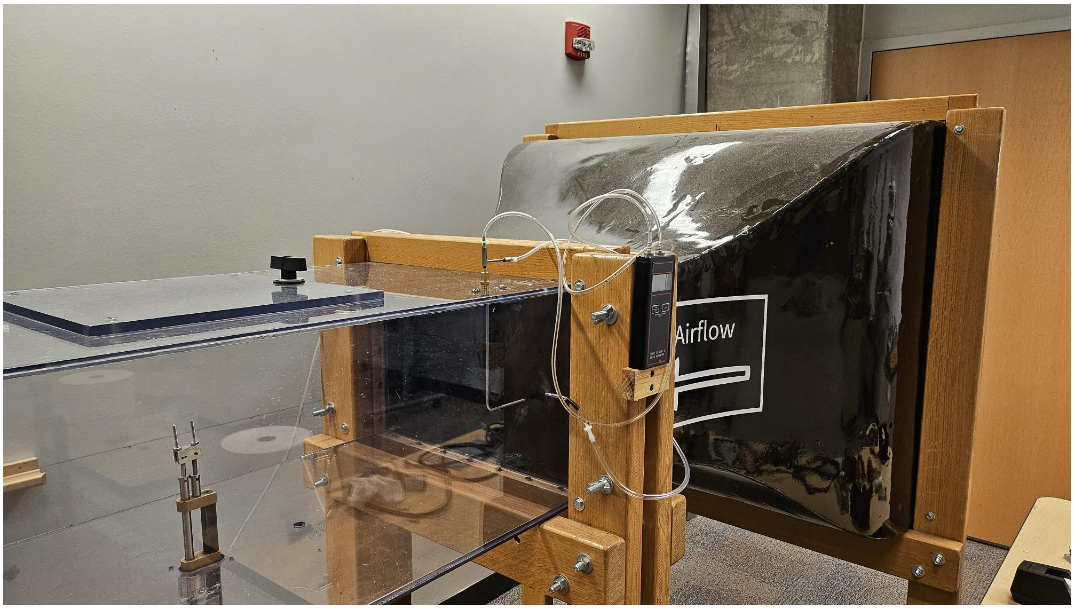


Figure 7.1: Armstrong Wind Tunnel at ARMS

Operation

1. Make sure Power Breaker (figure 7.2) is in "ON" position (red handle up). Do NOT move the handle if it is OFF, contact the lab technician.

Figure 7.2: Breaker for power control to wind tunnel (shown in ON Position)

2. Press the green “on” button (with vertical white line) on Allen Bradley controller (7.3)
3. Adjust speed (motor Hz) using up and down arrow buttons on the panel.
4. Press the red "off" (with white circle) button to turn off the wind tunnel.

7.1.1 Instrumentation

A pitot static probe inside the tunnel is used to measure the velocity of the wind tunnel. The pitot probe measures the static and total pressure, which is fed to a digital and an analog manometer. The manometers can subsequently be used to measure the dynamic pressure, to obtain the wind velocity. This wind velocity can be further correlated to the frequency of drive that controls the wind tunnel fans.

7.1.2 Force Balance Calibration

Platform Balance

The platform balance is used to measure the drag, lift and pitching moment produced on an airfoil in the wind tunnel. It consists of four FUTEK Force Transducers. Three of the transducers are located under the top balance plate and measure the lift and pitching moment. These are labeled F1, F2, and F3. The fourth transducer is located inside the parallelogram balance measuring the drag and is labeled F0. The transducers are all connected to the National Instruments Channel Amplifiers. A schematic of the set up is shown below in Figure 7.4.

The balance works by taking the readings from the different transducers and manipulating them with the proper equations to output the lift, drag and pitching moment. To calibrate the four transducers on the

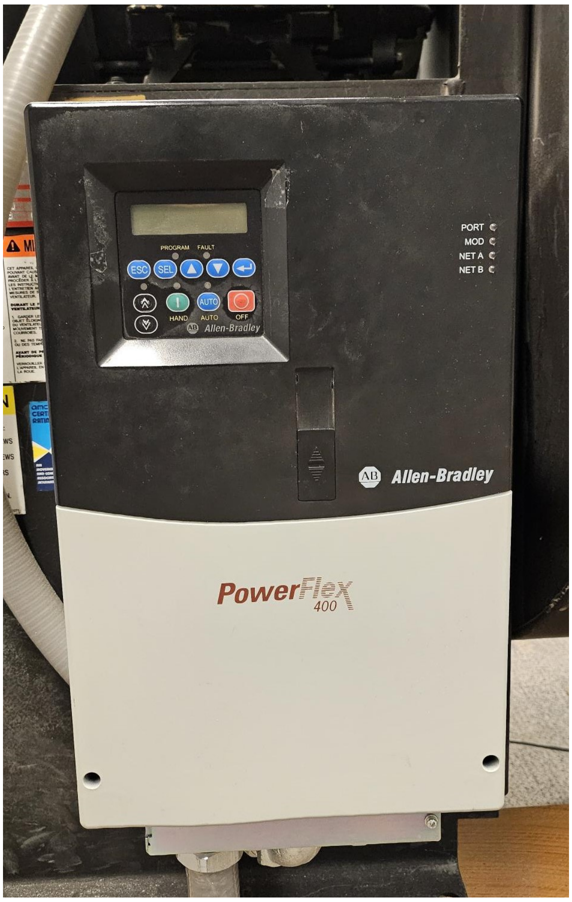


Figure 7.3: Allen Bradley Controller for controlling the variable frequency drive of the wind tunnel for speed control

platform balance, it is necessary to apply a known load at two different locations on the balance to measure the lift and drag.

- The Drag is measured by the F0 transducer reading inside the parallelogram. It measures the force induced between the top plate and the bottom plate. The drag force is given by equation 7.1.

$$D = F0; \tag{7.1}$$

- The Lift is measured by the F1, F2, and F3 transducers located beneath the top plate. When the airfoil is placed in the wind tunnel, the lift force pulls up on the balance. This Lift Force is the sum of the transducers. The force is given by the equation 7.2.

$$L = F1 + F2 + F3; \tag{7.2}$$

7.1.3 General process of calibration of force balance

This force balance needs to calibrated to get appropriate lift and drag readings. For calibrating lift, known weights should be placed on the **top balance plate** and for drag, put the weights **on the pulley platform** located just behind the F2 transducer. The procedure is as follows:

1. Make sure the tunnel is not running, contains no test specimen in the test section and that there are no weights placed on the force balance.
2. Run the WindTunnelBalance VI. When you run the program, it zeroes out all the forces in the sensors.
3. Record the initial values of lift/drag, against zero actual weight (when no weights are placed on the balance) from the VI.
4. Apply known weight(s) of 0, 0.5, 1, 2, 3, 4 and 5 lbs sequentially by stacking it based on which force (lift or drag) is being calibrated. Let the balance even out (also record the weight given by LabVIEW when no weights are placed). Place half pound and then the subsequent weights. Note that the curve is quite linear for most part, but remember that calibration is imperative to get accurate data.

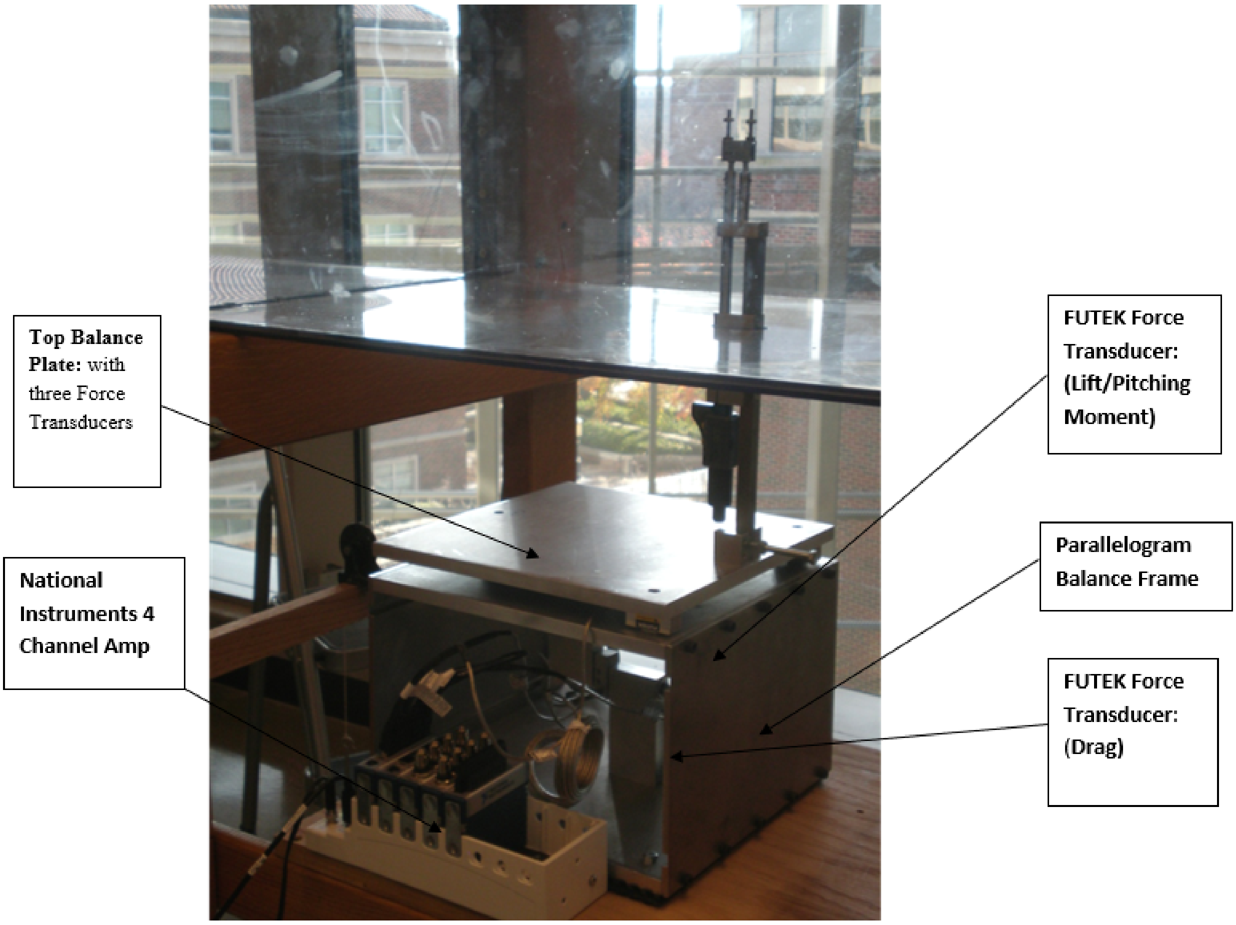


Figure 7.4: Force Balance Set-up

5. Record the values of lift/drag from the program agasint the actual weight.
6. Plot "Actual Weight for Lift or Drag" vs "LabVIEW Weight for Lift or Drag".

7.1.4 Angle of Attack Adjustment

The angle of attack of the airfoil is changed using a micrometer which is mounted on a metal stand, which in turn is screwed to the force balance platform. Figure 7.5 shows the various parts of the test stand along with the micrometer.

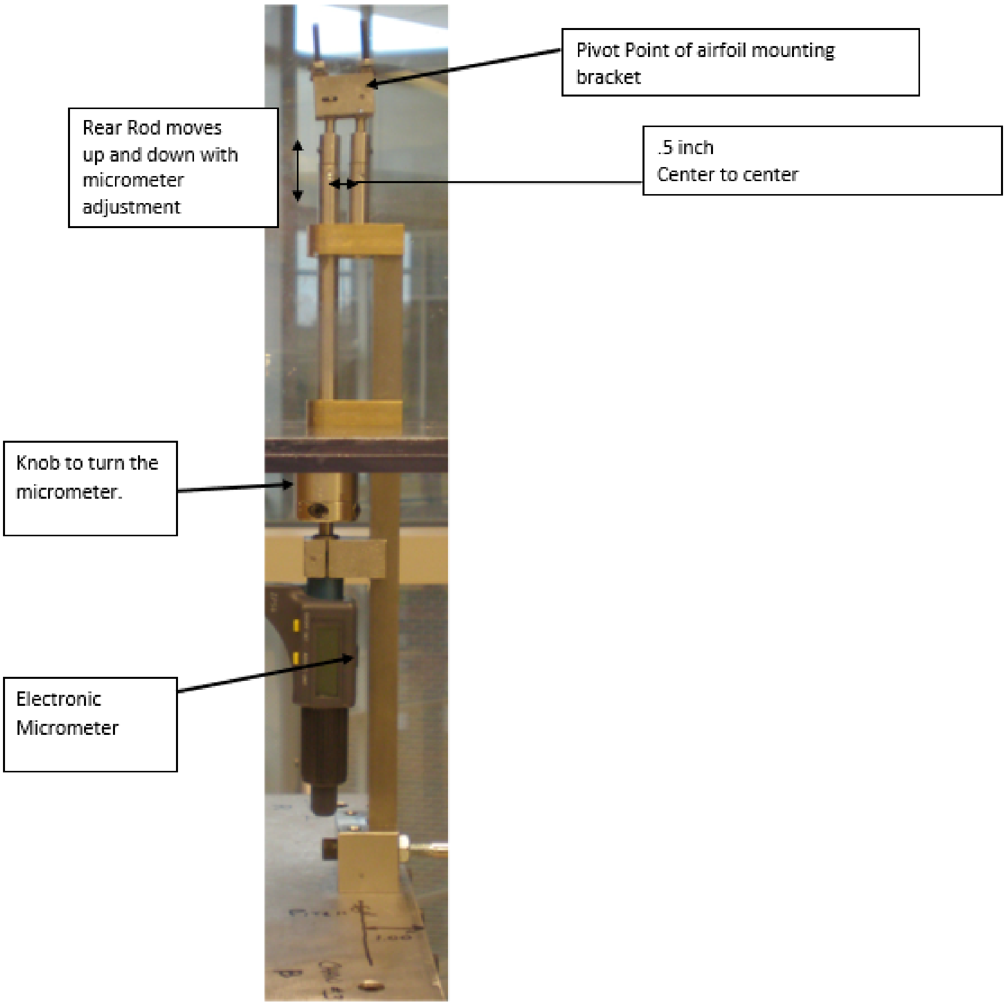


Figure 7.5: Anatomy of the airfoil test stand

Follow these steps after mounting your stand (with the micrometer) onto the force balance:

1. Mount the airfoil model on the stand using nuts and appropriately sized wrenches.
2. Use a level to zero out the angle of attack of the airfoil. The level must be set on top of the two mounting screws. Turn the knob to manipulate the angle of attack.
3. Once the angle of attack is zero (the level shows that it is perfectly horizontal), set micrometer value to zero by pressing ABS/INC button. This will "tare" the micrometer.
4. Use trigonometry to calculate the required micrometer value for a given angle of attack. Equation 7.3 can be used to find micrometer value in inches (h) corresponding to desired angle of attack (α). Note that 0.5 inch is the distance between the two mounting studs. Also, the h value obtained using this equation will be in inches, however the micrometer itself may show value in millimeters, please convert the value appropriately.

$$\boxed{h = 0.5 \times tan\alpha} \tag{7.3}$$

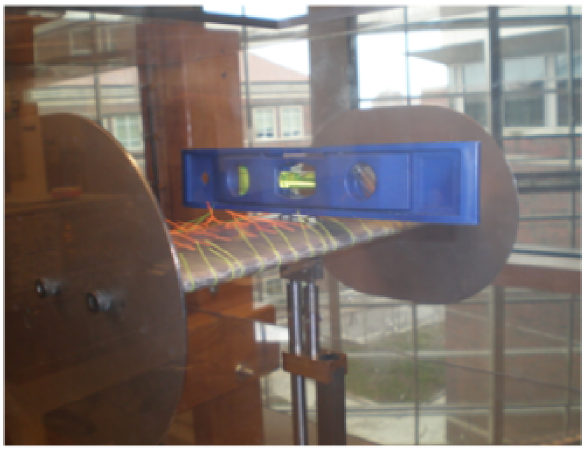


Figure 7.6: Level for Zeroing Angle of Attack

7.1.5 Micrometer Datasheet

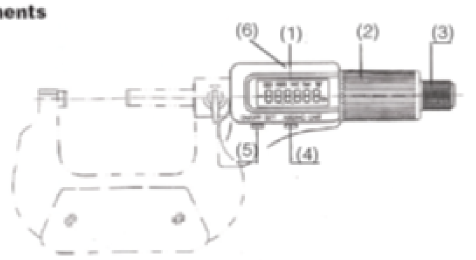
The micrometer mounted on the airfoil stand is modified to affect the angle of attack as the knob is rotated. Refer to the datasheet (7.7) of the instrument.

Fowler

Electronic Micrometer
Part Number: 54-850/860 series

Operation Manual

1. Functional elements



(1): LCD display
(2): Friction drive
(3): Quick drive
(4): ABS/INC•••UNIT key
(5): ON/OFF•••SET key
(6): Data output

2. Keys

ON/OFF•••SET key: Power switch. Datum set.
ABS/INC•••UNIT key: Absolute & relative measuring. Metric/Inch conversion.

3. LCD Display

"ABS" : Absolute measuring mode.
"INC" : Relative measuring mode.
"SET" : Datum set.
"in" : Unit: inch, otherwise mm.
: Battery voltage low.
: Data output is transmitting.

4. Operation

Two ways of pressing the keys are used in the following illustrations:
(1) Press and release; (2) Press and hold (2 sec. or more).

4.1 ON/OFF•••SET key:

Press and release: Power on/off.
Press and hold (2 sec. or more): Datum setting for absolute measurement; "Set" sign displayed on LCD.
Origin of metric is 0, 25, 50, 75 ••• 275mm. Origin of inch is 0, 1", 2", 3" ••• 11".
Sets datum automatically after battery reset.

4.2 ABS/INC•••UNIT key:

Press and release: Absolute and relative measuring mode conversion; "INC" sign displayed on LCD in relative measuring mode. "ABS" sign displayed on LCD in absolute measuring mode.
Press and hold (2 sec. or more): Metric/Inch conversion; "in" sign displayed on LCD for inch, otherwise mm.

5. Power

- A silver oxide cell SR44 is inserted into the back of the instrument with the positive pole outward . Replace the battery when the display data is dim or the " " sign is displayed in the upper left of the LCD.
- If not used for approximately five minutes, the power will auto-off. The micrometer will turn on by pressing "ON/OFF•••SET" key or turning the spindle. Power off the micrometer by pressing "ON/OFF•••SET" key to save the battery.

6. Specifications

Measuring force: 5 ~ 10N Power consumption: ≤ 20 μ A
Operating temperature: 0 ~ 40°C Storage temperature: -20 ~ 60°C

7. Data Output

1200 Baud, no parity, 7 data bits, 2 stop bits, no flow control.

8. Precautions

- Do not subject the instrument to blows or knocks. Do not drop it or apply excessive force.
- Do not disassemble the instrument.
- Do not press the keys with a pointed object.
- Do not use or store the instrument under direct sunlight, or in an excessively hot or cold area.
- Do not use the instrument near strong magnetic fields and high voltages.
- Use a soft cloth or a cotton swab that is dry to clean the instrument. Do not use organic solvent such as acetone or benzene. Alcohol may be used.
- Wipe the measuring faces of the instrument before using it.
- Remove the battery if the instrument is not used for a long period of time.

9. Troubleshooting

Failure	Causes	Repairing The Failure
Display "E 1" on LCD.	Data overflow.	Move spindle in reverse or press "ON/OFF•••SET" key.
Display "E 3" on LCD.	1. Sensor overflow. 2. Something wrong with sensor.	1. Reset battery (remove for 3 minutes). 2. Return the micrometer for repair.
Measuring data is not correct.	1. Dirty measuring surfaces. 2. Preset data is not correct.	1. Clean measuring surfaces. 2. Inspect preset data and reset it.
No display on LCD.	1. Battery voltage under 1.45v. 2. Battery is not properly set.	1. Replace battery. 2. Reset battery (remove for 3 minutes).
Display confused or remains blank.		Replace battery.
1. Display blurred. 2. The output data is wrong.	Battery voltage under 1.45v.	Replace battery.

Fred V. Fowler Co., Inc. • 66 Rowe Street • Newton, MA 02466
617-332-7004 • 617-332-4137 (fax) • Internet: www.fvfowler.com

Figure 7.7: Datasheet for Electronic Micrometer

7.2 Boeing Wind Tunnel

The Boeing Subsonic Wind Tunnel shown in Figure 7.8, located in the Aerospace Sciences Laboratory (ASL) at Purdue University, is a closed-return, closed test-section wind tunnel with a 4 ft. x 6 ft. x 8 ft.-long test section. With an empty test section, the wind tunnel is able to reach speeds of 96 mph (43 m/s). Students typically run the tunnel at 20 Hz, which yields an airspeed of about 13 m/s.

The tunnel is equipped with the following instrumentation:

1. Four component force balance
2. Lift, drag, pitch, roll
3. Computer-controlled angle of attack
4. Computer-driven two-axis traverse
5. 80 inch horizontal travel × 60 inch vertical travel probe
6. Pitot probes, seven hole probe, air velocity transducer, temperature transducer
7. Smoke wand
8. LabVIEW data acquisition system

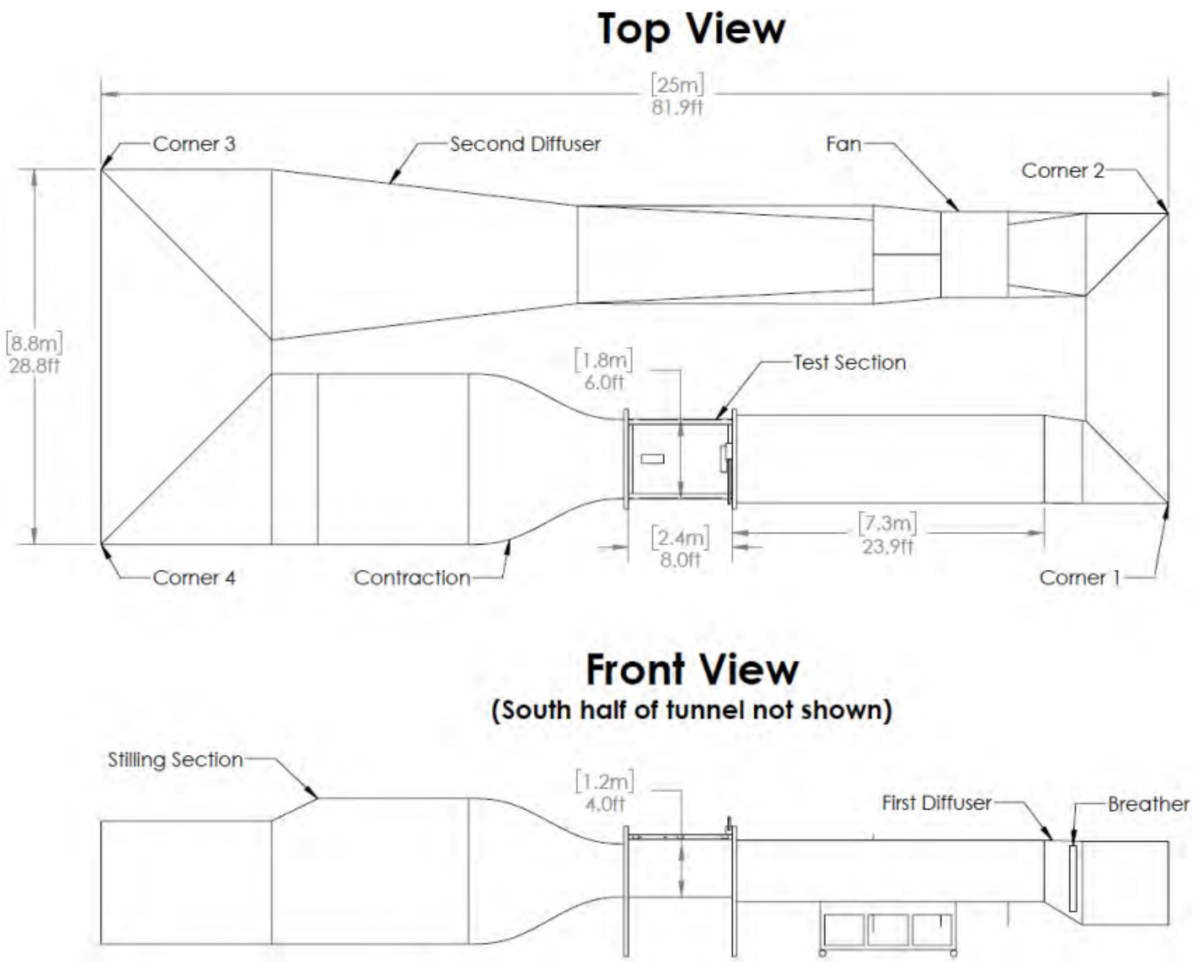


Figure 7.8: Boeing Subsonic Wind Tunnel Overview

This section is derived from the Boeing Subsonic Wind Tunnel Operation Manual. *Diffey, C. S. (2019). Characterization of The Flow Quality in the Boeing Subsonic Wind Tunnel (Version 1). Purdue University Graduate School. https://doi.org/10.25394/PGS.9105095.v1*

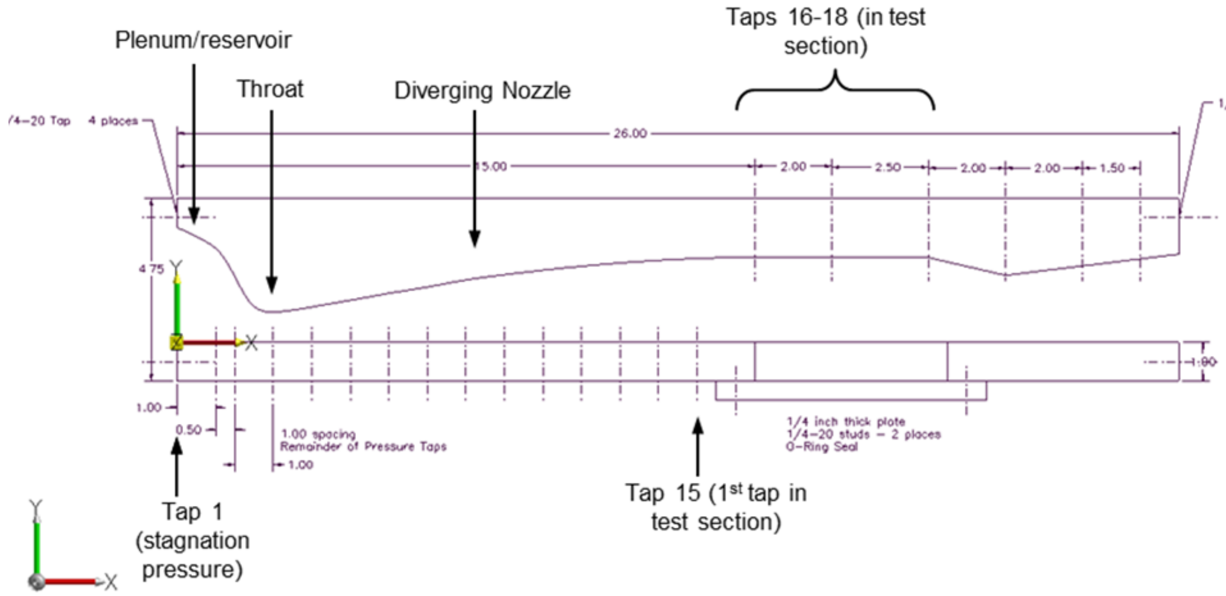


Figure 7.9: Diagram of the supersonic wind tunnel and locations of pressure taps

7.3 Supersonic Wind Tunnel A

7.3.1 Pressure Measurement and Pressure Scanner

1. Notice the pressure tap locations, starting behind the nozzle throat (location of minimum area) in the reservoir and distributed along the diverging nozzle. The first pressure tap is in the reservoir and so that measures the total (or stagnation) pressure p_0.
2. The pressure tap numbers, distance relative to the throat, and the corresponding pressure transducer channel number are shown in Figure 7.9 and Table 7.1. Note that the tap number does not necessarily correspond to the channel number; this is because some channels are low-pressure channels (Channel #1 to #13) and others are used to measure high pressures (Channel #14 to #16).
3. The nozzle height at each pressure tap location is also given in Table 7.1. The tunnel width is 1”, so the area at each tap location is simply the height multiplied by 1.
4. If for any reason tubing becomes disconnected, please get the TA to reattach the tubing. This is a trivially simple step compared to replacing the box if a high-pressure tube were to be mistakenly attached to a low-pressure transducer port. Please don’t hesitate to get the TA involved.

7.3.2 Supersonic Wind Tunnel Operation

The main features of the tunnel operation are shown in Figure 7.10. The tunnel is quite loud, and so it is extremely important that everyone in the room wears ear protection any time the tunnel is running. **Before you turn on the tunnel, you must verify that everyone in the room has on ear protection.**

Tap Number	**Distance from throat to pressure tap location (in)**	**Height of tunnel cross-section (in)**	**Pressure Scanner Channel No.**
1	-1.5	2.455	16
2	-0.5	0.937	15
3	0	0.796	14
4	1	0.926	13
5	2	1.1	12
6	3	1.281	11
7	4	1.447	10
8	5	1.628	9
9	6	1.767	8
10	7	1.883	7
11	8	1.986	6
12 (not used)	9	2.074	Not used
13	10	2.14	5
14	11	2.189	4
15	12	2.213	3
16	13	2.213	2
17	14	2.213	1

Table 7.1: Pressure tap information.

7.4 Supersonic Wind Tunnel B

Supersonic Tunnel B is used for analyzing the thrust produced by a converging diverging nozzle. Tunnel B is shown in 7.11. The tunnel is fully electronic and does not use an analog instrumentation for control. All data acquisition and control is done via the computer terminal installed right next to the tunnel. This tunnel is under beta testing: any feedback regarding the operation of tunnel, including cases where anomalistic behavior was seen can be reported to the course instructor for remidiation.

The LabVIEW interface for data acquisition and control of Tunnel B is seen in Lab 5 for AAE333401. The readouts include the reservoir pressure (the pressure from the supply line), the plenum pressure (pressure maintained inside the test section), pressure at various locations along the nozzle, the thrust as a result of the airflow.

To turn the tunnel on, press the Tunnel Switch, which will start the data acquisition and control systems for the tunnel. Use the Plenum pressure valve control slider or the precise plenum pressure control to give voltage to the valve. As the voltage is increased, the valve also opens more, allowing greater airflow. Stop increasing the voltage when the desired plenum pressure is achieved.

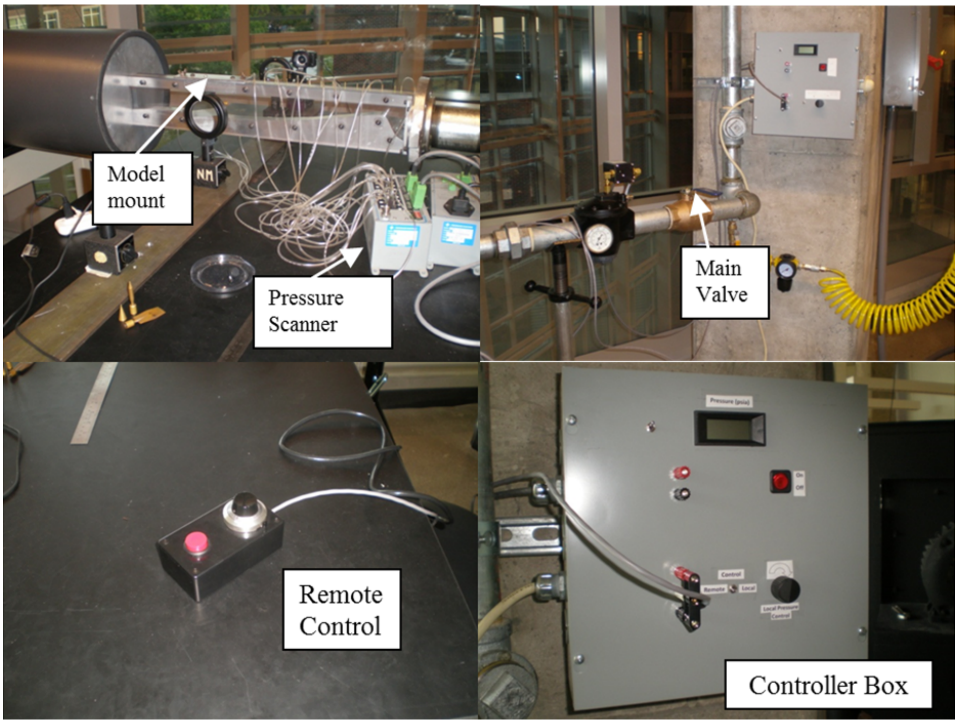


Figure 7.10: Supersonic wind tunnel and features for operation

Figure 7.11: Compressible Nozzle Experiment

7.5 Introduction to LabVIEW

LabVIEW (short for **Lab**oratory **V**irtual **I**nstrument **E**ngineering **W**orkbench) is a fully featured Integrated Development Environment (IDE) produced by National Instruments. It is a graphical programming environment designed for automated test and measurement. Engineers and scientists use LabVIEW in a wide array of applications and industries to integrate hardware and collect data in their test and measurement systems.

Here are some salient features of LabVIEW:

1. **Purpose-Built for Test and Measurement**
   - LabVIEW is specifically designed for creating test and measurement systems. It provides a unique approach to programming, connectivity to various instruments, and fully integrated user interfaces, via its graphical programming environment.
   - Engineers can use LabVIEW to build custom measurement and control applications, automate testing processes, and acquire data from sensors, instruments, and other hardware.
2. **Graphical Programming Environment**

- LabVIEW uses a graphical programming language called G. Despite its name, many people refer to the language as "LabVIEW."
- Instead of writing code in traditional text-based languages, LabVIEW users create programs by connecting graphical icons (nodes) that represent functions, data flow, and control structures.
- This visual approach makes it easier to understand and develop complex systems.

3. **Hardware Integration and Data Analysis**
   - LabVIEW simplifies hardware integration for engineering applications. It provides a consistent way to acquire data from National Instruments (NI) hardware as well as third-party devices.
   - The software supports drag-and-drop user interface (UI) creation and integrated data viewers, allowing engineers to immediately visualize results.
   - Engineers can also develop algorithms for data analysis and advanced control using built-in math and signal processing functions or by reusing their own libraries.

7.6 Numerical Methods

7.6.1 Trapezoidal Rule

The trapezoidal rule us a technique for numerical integration, that can be used to perform integration of discrete values over a given interval. Essentially, it can be used for calculating a definite integral $\int_a^b f(x)dx$.

For a non-uniform grid spacing, one can use the following formula.

$$\int_a^b f(x)dx \approx \sum_{k=1}^{N} \frac{f(x_{k-1}) + f(x)}{2} \Delta x_k \tag{7.4}$$

where $\Delta x_k = x_k - x_{k-1}$

Glossary

LabVIEW or **Lab**oratory **V**irtual **I**nstrument **E**ngineering **W**orkbench is a systems engineering software developed by National Instruments that is primarily used for data acquisition, instrument control and automation.. 4, 17