MIDTERMS

MODULE 1

(topic 1)

CONTINUITY EQUATION

The principle of mass conservation in fluid dynamics is governed by the continuity equation. The **continuity equation** is a fundamental principle in fluid mechanics that expresses the **conservation of mass** in a flow system. It states that for a fluid moving through a pipe, duct, or any control volume, the **mass flow rate remains constant** between any two points, assuming there are no leaks or additional mass sources.

The mass flow rate **($\dot{m}$)** represents the amount of mass passing through a given cross-section per unit time. It is defined as:

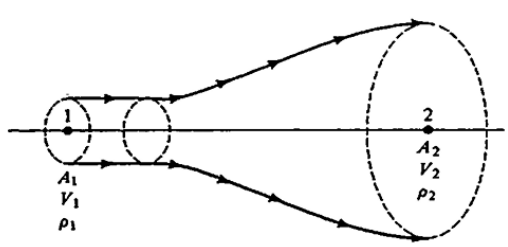
$$\dot{m} = \rho A V$$

where:

- **ρ = density of the fluid**
- **V = velocity of the fluid**
- **A = cross-sectional area of the flow**

Since mass must be conserved, the mass flow rate at one point must be equal to the mass flow rate at another point:

$$\dot{m}_1 = \dot{m}_2$$



Let:

A_1, A_2 = the cross sectional area of the stream tube at point 1 and 2
V_1, V_2 = flow velocity at point 1 and 2

This is a statement of the principle of mass conservation for a steady, one-dimensional flow, with one inlet and one outlet. This equation is called the continuity equation for steady one-dimensional flow. Since the mass at point 1 is the same as point 2.

For incompressible, the **density remains constant** ($\boldsymbol{\rho_1 = \rho_2}$), simplifying the equation to:

$$\boldsymbol{V_1 A_1 = V_2 A_2}$$

This means that if the cross-sectional area decreases, velocity increases and if the cross-sectional area increases, velocity decreases. If the flow velocity is equal to or below **100 $\boldsymbol{\frac{m}{s}}$ or 328.084 $\frac{ft}{s}$,** it is considered to be incompressible.

For incompressible flow, density of the fluid elements can change from point to point. It mjust be written in full equation as:

$$\boldsymbol{\rho_1 V_1 A_1 = \rho_2 V_2 A_2}$$

Sample Problem:

Consider a compressible convergent duct with an inlet area of 7 m^2 and an exit area of 3 m^2. The air enters inside the duct with a velocity of 500 $\frac{m}{s}$ and a density of 0.001 $\frac{kg}{m^3}$. The air leaves the duct at a velocity of 1000 $\frac{m}{s}$. What is the density at the exit?

Solution:

To solve for the **density at the exit** ($\boldsymbol{\rho_2}$) using the **continuity equation**, we will use the formula:

$$\boldsymbol{\rho_1 V_1 A_1 = \rho_2 V_2 A_2}$$

Given:

$\boldsymbol{\rho_1}$ = Inlet density = 0.001 $\frac{kg}{m^3}$

$\boldsymbol{V_1}$ = Inlet velocity = 500 $\frac{m}{s}$

$\boldsymbol{A_1}$ = Inlet area = 7 m^2

$\boldsymbol{V_2}$ = Exit velocity = 1000 $\frac{m}{s}$

$\boldsymbol{A_2}$ = Exit area = 3 m^2

$\boldsymbol{\rho_2}$ = Exit density =?

Step-by- Step:

1. Rearrange the continuity equation to solve for $\boldsymbol{\rho_2}$:

$$\boldsymbol{\rho_1 V_1 A_1 = \rho_2 V_2 A_2 \longrightarrow \rho_2 = \frac{\rho_1 V_1 A_1}{V_2 A_2}}$$

2. Substitute the given values:

$$\boldsymbol{\rho_2 = \frac{(0.001)(500)(7)}{(1000)(3)}}$$

3. Calculate the values:

$$\rho_2 = \frac{0.001 \times 3500}{3000}$$

$$\rho_2 = \frac{3.5}{3000}$$

$$\rho 2 = 0.001167 \frac{kg}{m^3}$$

4. The density at the exit $\boldsymbol{\rho_2}$, is approximately **0.001167** $\frac{kg}{m^3}$

(topic2)

SPEED OF SOUND

The sound waves that travel through the air at definite speed is called **speed of sound**. The **speed of sound** is the rate at which this disturbance propagates through the medium. It is the velocity at which the pressure wave moves, not the velocity at which the individual particles of the medium move. The speed of sound in a perfect gas depends only on the temperature of the gas. The standard sea level condition of speed of sound is approximately **1116** $\frac{ft}{s}$ **or 340** $\frac{m}{s}$.

The speed of sound is express as:

$$\boldsymbol{a = \sqrt{\gamma R T}}$$

Where:

a = speed of sound

γ = ratio of specific heat with a constant value of **1.4**

R = Universal Gas Constant with a value of **287** $\frac{J \cdot K}{kg}$ **or 1716** $\frac{ft \cdot lb}{slug} \cdot R$

T = temperature

Sample Problem:

Calculate the speed of sound at an altitude of 5km.

Given:

- **h = 5 km**

Solution:

Step 1: Convert 5 km to meters

$$5 \text{ km} \times \frac{1000 \text{ m}}{1 \text{ km}} = 5000 \text{ m}$$

Step 2: Using the formula of getting the temperature of a gradient layer, solve for the temperature at 5000 m.

Formula:

$$T = T_0 + (a \cdot h)$$

$$T = 288.15K + (-0.0065 \text{ K/m} \times 5000 \text{ m})$$

$$T = 255.7K$$

Step 3: Solve for the speed of sound using the formula

Formula:

$$a = \sqrt{\gamma R T} a$$

$$a = \sqrt{(1.4)(287)(255.7K)}$$

$$a = 320.5312 \, m/s$$

Final Answer:
The speed of sound at 5 km is 320.5312 m/s.

(topic 3)

Mach Number

The ratio of the speed of the aircraft to the speed of sound in the gas determines the magnitude of many of the compressibility effects. Because of the importance of this speed ratio, aerodynamicists have designated it with a special parameter called the **Mach number** in honor of **Ernst Mach**, a late 19th century physicist who studied gas dynamics. The Mach number **M** allows to define flight regimes in which compressibility effects vary.

The formula for Mach Number is:

$$\boldsymbol{M} = \frac{\boldsymbol{V}}{\boldsymbol{a}}$$

where:

$\boldsymbol{M}$ **=** Mach Number

$\boldsymbol{a}$ **=** speed of sound

$\boldsymbol{V}$ **=** velocity of the object

Mach Number Conditions

0-0.8 = Subsonic

0.81- 1.2 = Transonic

1= Sonic

1.21- 5.0= Supersonic

5.1- 25.0 = Hypersonic

>25.1 = Ultrasonic

Sample Problem:

An aircraft is cruising at an altitude of 4000 meters with a velocity of 217.30 m/s. As part of an aircraft performance analysis, the pilot wants to determine the Mach number of the aircraft at this altitude.

Given:

- h = 4000 m
- V = 217.30 m/s
- M = ?

Step 1: First solve the temperature at 4000 m

Formula:

$$T = T0 + (a \cdot h)$$

$$T = 288.2K + (-0.0065\ K/m \times 4000\ m)$$

$$T = 262.2K$$

Step 2: Solve for the speed of sound

Formula:

$$a = \sqrt{\gamma RT}$$

$$a = \sqrt{(1.4)(287)(262.2K)}$$

$$a = 324.579666\, m/s$$

Step 3: Solve the Mach Number using the formula

$$M = \frac{V}{\alpha}$$

$$M = \frac{217.30\, m/s}{324.5790666\, m/s}$$

$$M = 0.67\ (Subsonic)$$

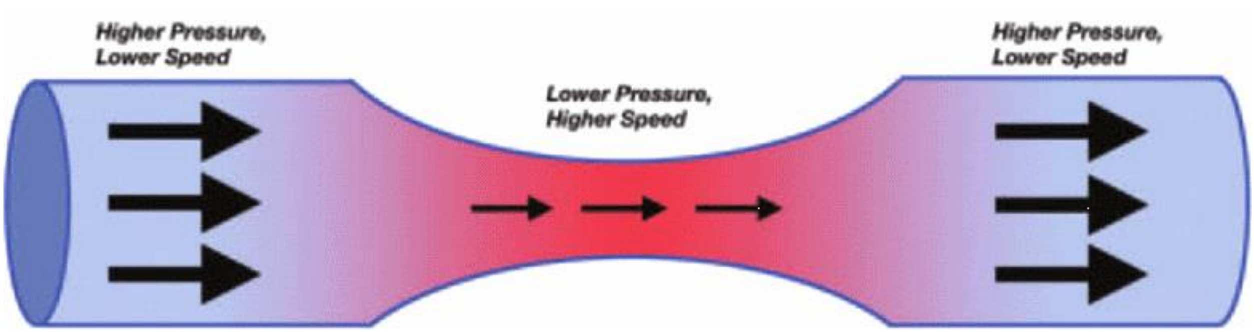
Final Answer:
The Mach Number at 4000 m with a velocity of 217.30 m/s is **0.67 subsonic**.

(topic 4)

Bernoulli's Equation

The Bernoulli Equation can be a statement of the conservation of energy principle appropriate for flowing fluids. The qualitative behavior that is usually labeled with the term "Bernoulli effect" is the lowering of fluid pressure in regions where the flow velocity is increased. This lowering of pressure in a constriction of a flow path may seem counterintuitive but seems less so when you consider pressure to be energy density. In the high velocity flow through the constriction, kinetic energy must increase at the expense of pressure energy

Bernoulli effect simply states that **an increase in velocity** leads to **decrease in pressure**. The higher the velocity, the lower the pressure.



If the flow is incompressible, the Bernoulli's equation is defined as:

$$\boldsymbol{P_1 + \rho\frac{V_1^2}{2} = P_2 + \rho\frac{V_2^2}{2}}$$

Where:

$\boldsymbol{P_1}$ = Pressure at the inlet

$\boldsymbol{\rho_1}$ = Density at the inlet

$\boldsymbol{V_1}$ = Velocity of the flow at the inlet

$\boldsymbol{P_2}$ = Pressure at the exit

$\boldsymbol{\rho_2}$ = Density at the exit

$\boldsymbol{V_2}$ = Velocity of the flow at the exit

If the flow is compressible, the Bernoulli's equation is defined as:

$$\boldsymbol{P_1 + \rho_1\frac{V_1^2}{2} = P_2 + \rho_2\frac{V_2^2}{2}}$$

Where:

$\boldsymbol{P_1}$ = Pressure at the inlet

$\boldsymbol{\rho_1}$ = Density at the inlet

$\boldsymbol{V_1}$ = Velocity of the flow at the inlet

$\boldsymbol{P_2}$ = Pressure at the exit

$\boldsymbol{\rho_2}$ = Density at the exit

$\boldsymbol{V_2}$ = Velocity of the flow at the exit

derivations:

When finding the pressure, we can use the formula:

$$\boldsymbol{P_1 = P_2 + \rho\frac{V_2^2}{2} - \rho\frac{V_1^2}{2}}$$

$$\boldsymbol{P_2 = P_1 + \rho\frac{V_1^2}{2} - \rho\frac{V_2^2}{2}}$$

When finding the velocity, we can use the formula:

$$\boldsymbol{v_2 = \sqrt{\frac{2\left(P_1 + \frac{\rho v_1^2}{2} - P_2\right)}{\rho}}}$$

$$\boldsymbol{v_1} = \sqrt{\frac{2\left(\boldsymbol{P_2} + \frac{\boldsymbol{\rho v_2^2}}{\boldsymbol{2}} - \boldsymbol{P_1}\right)}{\boldsymbol{\rho}}}$$

Sample Problem:

An aircraft engine duct is designed to gradually slow down the airflow as it moves through. Air enters the duct with a velocity of 150.54 m/s at an inlet pressure of 140,516 Pa. At the duct exit, the flow has a Mach number of 0.5 and a temperature of 277 K. The flow is considered incompressible, with a constant air density of 1.07 kg/m^3. Find the pressure at the exit.

Given:

V_1 = 150.54 m/s
P_1 = 140,516 Pa
M_2 = 0.5
T_2 = 277K
ρ = 1.07 kg/m^3
P_2 =?

Step 1: To find the velocity at the exit, solve for the speed of sound using the formula:

$$a = \sqrt{\gamma RT}$$

$$a = \sqrt{(1.4)(287)(277)}$$

$$a = 333.6144481 \text{ m/s}$$

Step 2: Now derive the Mach number formula to get the velocity at the exit:

$$M = \frac{v}{a}$$

$$V = (M)(a)$$

Step 3: Solve for V_2

$$V2 = (M)(a)$$

$$V2 = (0.5)(333.6144481)$$

$$V2 = 166.8072241\ m/s$$

Step 4: Since the condition is incompressible with a density of 1.07 kg/m^3, we can now solve for the value of $P2$ using Bernoulli's equation:

$$P1 + \frac{\rho V_1^2}{2} = P2 + \frac{\rho V_2^2}{2}$$

Derived:

$$P_2 = P_1 + \frac{\rho V_1^2}{2} - \frac{\rho V_2^2}{2}$$

Step 5: Substitute the values

$$P_2 = (190.516\ \text{Pa}) + \frac{(1.07\ \text{kg/m}^3)(150.54\ \text{m/s})^2}{2} - \frac{(1.07\ \text{kg/m}^3)(166.8072241\ \text{m/s})^2}{2}$$

$$P2 = 137{,}754.1382\ Pa$$

MIDTERM

MODULE 2

(topic 1**)**

WIND TUNNEL

Wind tunnels are large tubes with air moving inside. The tunnels are used to copy the actions of an object in flight. Researchers use wind tunnels to learn more about how an aircraft will fly. NASA uses wind tunnels to test scale models of aircraft and spacecraft. Some wind tunnels are big enough to hold full-size versions of vehicles. The wind tunnel moves air around an object, making it seem like the object is really flying.

The object to be tested inside a wind tunnel is fastened in the tunnel so that it will not move. The object can be a small model of a vehicle. It can be just a piece of a vehicle. It can be a full-size aircraft or spacecraft. It can even be a common object like a tennis ball. The air moving around the still object shows what would happen if the object were moving through the air.

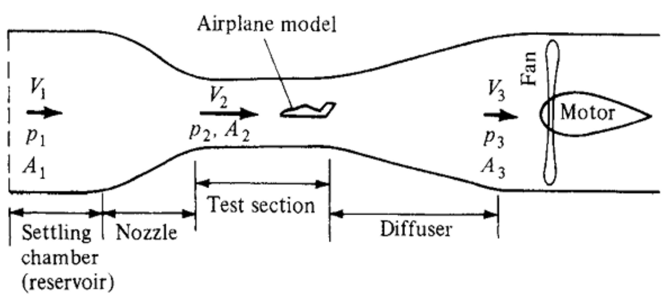
Wind Tunnels are usually made out of steel or aluminum, that are tested and loaded with many instruments and sensors that report back to the computers in the control room. In **wind tunnel testing** for aircraft, various forces and motions are studied to understand how the aircraft will behave under real flight conditions. The aerodynamic forces and motions in wind tunnel testing give analysis for improving aircraft design, performance, and safety. The forces and motions of aircraft under wind tunnel testing are **lift, drag, side force, pitching moment, yawing moment and rolling moment**.

TYPES OF WIND TUNNELS

1. Low-Speed Wind Tunnel

A **low-speed wind tunnel** is a type of wind tunnel designed to simulate airflow at speeds that are significantly slower than the speed of sound. These tunnels typically operate at speeds below **Mach 0.3**, where the flow remains **subsonic**. They are also used for **aerodynamic studies** on various models, such as **airfoils**, **wings**, **tail sections**, and **other vehicle components**. At these low speeds, the airflow is **smooth** (laminar), and the **compressibility effects** are negligible (meaning changes in air density due to pressure variations are minimal). The focus is often on **lift, drag, and moment measurements**, which are critical for understanding how the object behaves in steady flight conditions.

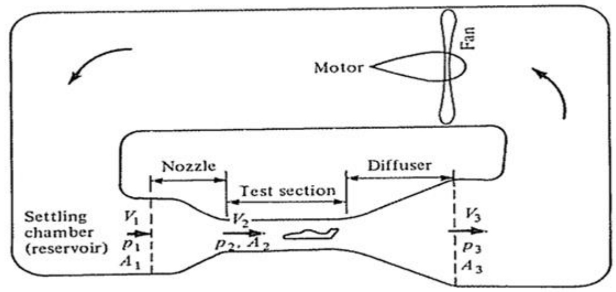
a. Open-circuit wind tunnel



(a) Open-circuit tunnel

An **open-circuit wind tunnel** is a type of wind tunnel in which the airflow is drawn from the surrounding environment and then expelled back into the atmosphere after passing through the test section. In this type of system, there is no recirculation of air; once the air flows through the tunnel, it exits and is replaced by fresh air from outside.

b. Closed-circuit low speed wind tunnel



(b) Closed-circuit tunnel

Closed-circuit wind tunnel also called (Eiffel or NPL) tunnel, is when the air flowing follows an essentially straight path from the entrance through a contraction to the test section, followed by a diffuser, a fan section, and an exhaust of the air. The air flowing recirculates continuously with little or no exchange of air with the exterior.

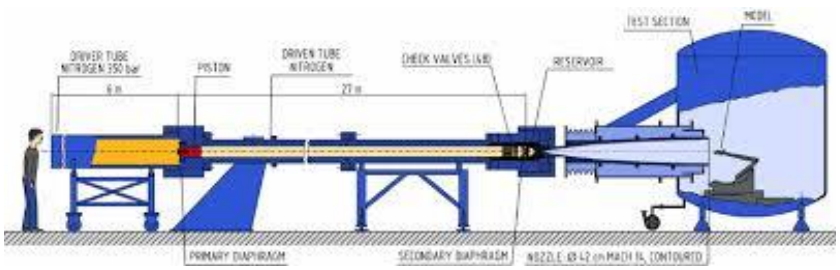
2. Transonic Wind Tunnel

Transonic wind tunnels are able to achieve speeds close to the speed of sound. The highest speed is reached in the test section. Testing at transonic speeds presents additional problems, mainly due to the reflection of the shock waves from the walls of the test section.

3. Supersonic Wind Tunnel

Supersonic wind tunnels should produce supersonic speeds (Mach numbers up to 5). This can be achieved with an appropriate design of a convergent divergent nozzle. When the sonic speed is reached in throat, the flow accelerates in a nozzle supersonically.

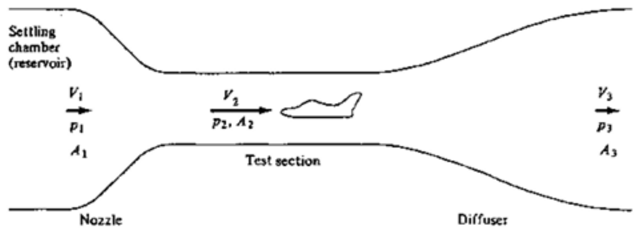
4. **Hypersonic Wind Tunnel**
   A hypersonic wind tunnel is designed to generate a hypersonic flow field in the working section. The speed of these tunnels varies from Mach 5 to 15. As with supersonic wind tunnels, these types of tunnels must run intermittently with very high-pressure ratios when initializing.



Aerodynamicists use wind tunnels to test models of proposed aircraft. In the tunnel, the engineer can carefully control the flow conditions which affect the forces on the aircraft. By making careful measurements of the forces on the model, the engineer can predict the forces on the full-scale aircraft. And by using special diagnostic techniques, the engineer can better understand and improve the performance of the aircraft.

The amount of air in the tunnel is a **constant**, and we can use the **conservation of mass** to relate local speed in the tunnel to the cross-sectional area. At every point in the tunnel, the velocity V times the air density ρ times the area A is a constant.

WIND TUNNEL ANALYSIS



A simple schematic of a subsonic wind tunnel

In wind analysis, particularly in wind tunnels or aerodynamic testing, Bernoulli's equation is often used to analyze airflow behavior at three key sections: **the inlet, the test section, and the exit**. The equation is:

$$\boldsymbol{P_1 + \rho_1 \frac{V_1^2}{2} = P_2 + \rho_2 \frac{V_2^2}{2} = P_3 + \rho_3 \frac{V_3^2}{2}}$$

Where:

$\boldsymbol{P_1}$ **=** Pressure at the inlet

$\boldsymbol{\rho_1}$ **=** Density at the inlet

$\boldsymbol{V_1}$ **=** Velocity of the flow at the inlet

$\boldsymbol{P_2}$ **=** Pressure at the test section

$\boldsymbol{\rho_2}$ **=** Density at the test section

$\boldsymbol{V_2}$ **=** Velocity of the flow at the test section

$\boldsymbol{P_3}$ **=** Pressure at the exit

$\boldsymbol{\rho_3}$ **=** Density at the exit

$\boldsymbol{V_3}$ **=** Velocity of the flow at the exit

Sample problem:

The airflow in the tunnel is compressible, and measurements are taken at different sections.At the intake, the airflow has a density of 0.98 kg/m^3 and a pressure of 320,153 Pa with 0.11 m/s of velocity. The mass flow rate in the tunnel is 0.72 kg/s. The density at the test section is recorded at 0.55 kg/m^3.The ratio of the intake area to the test section area is 0.73. Meanwhile, at the exit section, the area is ¾ size of the intake area, and the pressure at the exit is measured as 288,420.56 Pa.

Given:

$\rho 1 = 0.98 kg/m3$

$P1 = 320{,}553 Pa$

$V1 = 0.11 m/s$

$\dot{m} = 0.72 kg/s$

P_3 = 288,420.56 Pa.

$A_1 =?$

$A_2 =?$

V_3 = ?

Solution:

Step 1:

First, find the value of the area at the intake (A_1) using the mass flow rate formula:

$$\dot{m} = \rho A V$$

Derived formula:

$$A_1 = \frac{\dot{m}}{\rho_1 V_1}$$

Step 2:

Substituting the values:

$$A_1 = \frac{0.72 \text{ kg/s}}{(0.98 \text{ kg/m}^3)(0.11 \text{ m/s})}$$

$$A_1 = 6.67903525 m^2$$

Step 3:
Solve for the value of the area at the test section (A_2) using the formula

$$\frac{A_1}{A_2} = 0.73 \rightarrow A_2 = (0.73)\,(A_1)$$

Step 4:
Substituting the values:

$$A_2 = (0.73)(6.67903525\, m2)$$

$$A_2 = 4.875695733 m^2$$

Step 5:

Now that there is a value of A_1 solve for the value of the area at the exit (A_3):

$$A_1 \times \frac{4}{3} = A_3$$

$$A_3 = 6.7903526 \text{ m}^2 \times \frac{4}{3}$$

$$A3 = 5.009276438m2$$

Step 6:

Solve for the velocity at the test section (V_2)using the mass flow rate:

$$\dot{m} = \rho_2 A_2 V_2$$

$$V_2 = \frac{\dot{m}}{\rho_2 A_2}$$

Step 7:

Substitute the values:

$$V_2 = \frac{\dot{m}}{\rho_2 A_2}$$

$$V_2 = \frac{0.72 \text{ kg/s}}{(0.55 \text{ kg/m}^3)(4.875695733 \text{ m}^2)}$$

$$V_2 = 0.2684931507 \text{ m/s}$$

Step 8:

To get the value of pressure at the test section, we can use the Bernoulli equation:

$$P_1 + \frac{\rho_1 V_1^2}{2} = P_2 + \frac{\rho_2 V_2^2}{2} = \text{P}_3 + \frac{\rho_3 V_3^2}{2}$$

Step 9:

Since we only have the values at the intake and test section, we will use the formula to find P_2

$$P_2 = P_1 + \frac{\rho_1 V_1^2}{2} - \frac{\rho_2 V_2^2}{2}$$

$$P_2 = 320{,}153 \text{ Pa} + \frac{(0.98 \text{ kg/m}^3)(0.11 \text{ m/s})^2}{2} - \frac{(0.55 \text{ kg/m}^3)(0.2684931507 \text{ m/s})^2}{2}$$

$$P_2 = 320{,}152.9861\ Pa$$

Step 10:

To get the value of temperature at the exit, we can use the formula:

$$P_3 = P_0 \left[\frac{T_3}{T_0}\right]^{5.26}$$

Derived:

$$T_3 = T_0 \left[\frac{P_3}{P_0}\right]^{\frac{1}{5.26}}$$

Step 11:

Substitute the values

$$T_3 = 288.2\ \left[\frac{288{,}420.56}{101325}\right]^{\frac{1}{5.26}}$$

$$T_3 = 351.61275K$$

Step 12:

Using the temperature at the exit, solve for ρ_3 using the formula:

$$\rho_3 = \rho_o \left(\frac{T}{T_o}\right)^{4.26}$$

Step 13:

Substitute the values:

$$\rho_3 = 1.225 \text{ kg/m}^3 \times \left(\frac{288.2 \text{ K}}{351.61275 \text{ K}}\right)^{4.26}$$

$$\rho 3 = 2.858084371\ kg/m3$$

Step 14:

To solve for V_3, use the formula for mass flow:

$$\dot{m} = \rho_3 A_3 V_3$$

Derived:

$$V_3 = \frac{\dot{m}}{\rho_3 A_3}$$

Step 15:

Substitute the values:

$$V_3 = \frac{0.72\ \mathrm{kg/s}}{(2.858084371\ \mathrm{kg/m^3})(5.0092764380\ \mathrm{m^2})}$$

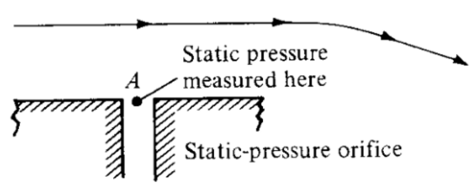
$$V3 = 0.05039020892\ m/s$$

MIDTERMS

MODULE 3

Static Pressure

Static pressure is a fundamental concept in fluid dynamics that refers to the pressure exerted by a gas due to the random motion of its molecules. It is the force per unit area that results from molecular collisions within the gas, independent of any bulk motion. This means that even if a gas is flowing at high speed, its static pressure is determined solely by the random movement of its particles, not by the overall velocity of the flow. Since static pressure is a direct result of molecular collisions, it is influenced by factors such as **temperature and density**. When **temperature increases**, molecules move faster and collide more forcefully, leading to **higher static pressure**. Similarly, an increase in gas density means more molecules are present in each volume, resulting in more frequent collisions and, consequently, greater pressure.



Total/Stagnation Pressure

Total pressure at a given point in a flow is the pressure that would exist if the flow were slowed down isentropically to zero velocity. At a stagnation point, the fluid velocity is zero and all kinetic energy has been converted into pressure energy. **Stagnation pressure** is equal to the sum of the free stream dynamic pressure and free-stream static pressure.

Pitot Tube

The **Pitot tube** is an aerodynamic instrument used to measure the **total pressure** at a point in a fluid flow. It was invented by the **French engineer Henri Pitot** in the early 18th century and has since become a fundamental tool in aerodynamics, aviation, and fluid dynamics. The primary function of a Pitot tube is to determine the **fluid flow velocity** by measuring the difference between total pressure and static pressure.

A Pitot tube operates based on **Bernoulli's principle**, which states that the total pressure in a steady, incompressible flow is the sum of static pressure (due to random molecular motion) and dynamic pressure (associated with the motion of the fluid). The instrument consists of a small, hollow tube with an opening facing directly into the flow. When air or fluid enters the tube, it comes to a stop (stagnation point), allowing the instrument to measure **total pressure** at that point.

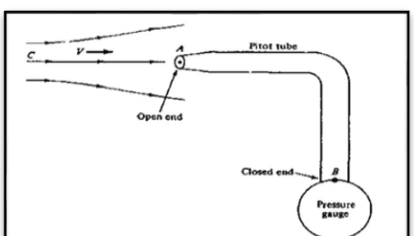


Figure 4.16 Sketch of a Pitot tube.

It consists of a tube placed parallel to the flow and open to the flow at one end (point A). The other end of the tube (point B) is closed. Any point of the flow where V=0 is called a **stagnation point.**

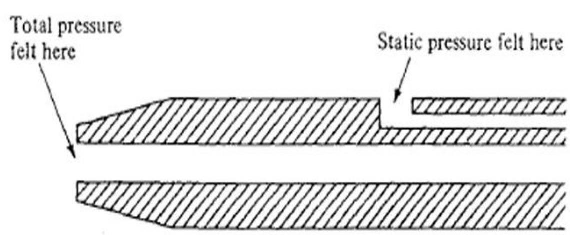
Static Port

The static pressure is obtained through a static port. The static port is most often a flush-mounted hole on the fuselage of an aircraft and is located where it can access the air flow in a relatively undisturbed area. Some aircraft may have a single static port, while others may have more than one. In situations where an aircraft has more than one static port, there is usually one located on each side of the fuselage. With this positioning, an average pressure can be taken, which allows for more accurate readings in specific flight situations.

PITOT-STATIC PROBE

The **Pitot-static probe** is an instrument used in aerodynamics and fluid mechanics to measure airspeed and fluid flow velocity. It is a specialized device that combines the functions of a **Pitot tube**, which measures **total pressure**, and a **static port**, which measures **static pressure**. This dual-function probe is widely used in aviation, wind tunnel testing, and various engineering applications where accurate airflow measurements are required.

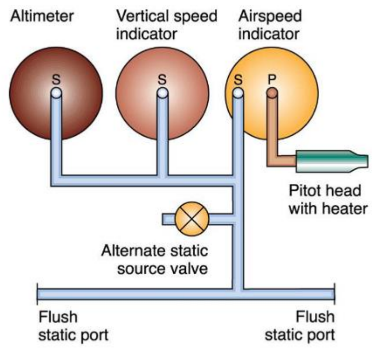
The **Pitot-static probe operates based on Bernoulli's principle**, which states that in a steady, incompressible flow, the total pressure of a fluid is the sum of its static pressure and dynamic pressure. The **total pressure** is captured by the Pitot tube, which has an opening facing directly into the airflow. At this point, the air is brought to a stop, creating a stagnation point where the total pressure can be measured. Meanwhile, the **static pressure** is measured through ports located on the sides of the probe, positioned perpendicular to the airflow to avoid the effects of fluid motion.



Pitot-static probe

PITOT-STATIC SYSTEM

A pitot-static system is a system of pressure sensitive instruments that is most often used in aviation to aircraft's airspeed, Mach determine a number, altitude, and altitude trend. A pitot-static system generally consists of a pitot tube, a static port, and the pitot-static instruments. Errors in pitot-static system readings can be extremely dangerous as the information obtained from the pitot static system, such as altitude, is often critical to a successful flight. Several commercial airline disasters have been traced to a failure of the pitot-static system



AIRSPEED

Airspeed is the speed of an aircraft relative to the air. Among the common conventions for qualifying airspeed are: **Indicated airspeed (IAS), Calibrated airspeed (CAS), True airspeed (TAS), Equivalent airspeed (EAS)** and **density airspeed.**

INDICATED AIRSPEED

Indicated airspeed (IAS) is the airspeed read directly from the airspeed indicator on an aircraft, driven by the pitot static system uncorrected for instrument and position errors.

CALIBRATED AIRSPEED

Calibrated airspeed (CAS) is the result of correcting indicated airspeed for errors in the measurement of the static pressure. **Calibrated airspeed (CAS)** is the IAS corrected for instrument and position errors.

$$V_c = V_I + \Delta V_p$$

Where:

V_c = calibrated airspeed

V_I = indicated airspeed

ΔV_P = calibration correction (read from the correction chart)

TRUE AIRSPEED

True Airspeed (TAS) also known as **knots true airspeed (KTAS)** of an aircraft is the speed of the aircraft relative to the air mass in which it is flying. It is the equivalent airspeed corrected for non-standard density (actual value of air density)

$$V_{TRUE} = \sqrt{\frac{2(P_T - P)}{\rho_{(actual)}}}$$

V SPEED

In aviation, **V-speeds** or **Velocity Speeds** are standard terms used to define airspeeds important or useful to the operation of aircraft, such as airplanes, gliders, autogiros, helicopters, blimps and dirigibles. These speeds are derived from data obtained by aircraft designers and manufacturers during flight testing and verified in most countries by government flight inspectors during aircraft type-certification testing. Using them is considered a best practice to maximize aviation safety, aircraft performance or both.

There are common V-speeds associating to an aircraft:

$\boldsymbol{V_A}$- design maneuvering speed (stalling speed at the maximum legal G force, and hence the maximum speed at which abrupt, full deflection, elevator control input will not cause the aircraft to exceed its G-force limit).

$\boldsymbol{V_{FE}}$- maximum flap extended speed

$\boldsymbol{V_{LE}}$- maximum landing gear extended speed. The maximum speed at which the aircraft may be flown with the landing gear extended.

$\boldsymbol{V_{LO}}$ - maximum landing gear operating speed. The maximum speed at which the aircraft may be flying while raising or lowering the gear.

$\boldsymbol{V_{MC}}$ or $\boldsymbol{V_{MCA}}$- - minimum control speed with the critical engine inoperative.

$\boldsymbol{V_{NE}}$ - the$\boldsymbol{V_{NE}}$, or never exceed speed, is the V speed which refers to the velocity that should never be exceeded because of the risk of structural failure, due for example to wing or tail deformation, or aeroelastic flutter.

$\boldsymbol{V_R}$- rotation speed. The speed of an aircraft at which the pilot initiates rotation to obtain the scheduled takeoff performance.

$\boldsymbol{V_S}$- the stalling speed or the minimum steady flight speed at which the aircraft is controllable.

$\boldsymbol{V_{S0}}$ - the stalling speed or the minimum steady flight speed in the landing configuration.

$\boldsymbol{V_{S1}}$ - the stalling speed or the minimum steady flight speed obtained in a specific configuration (usually a "clean" configuration without flaps, landing gear and other sources of drag).

$\boldsymbol{V_X}$ - speed for best angle of climb. This provides the best altitude gain per unit of horizontal distance and is usually used for clearing obstacles during takeoff.

$\boldsymbol{V_Y}$ - speed for best rate of climb. This provides the best altitude gain per unit of time.

Sample Problem:

An aircraft is flying at an altitude of 6,523 m. The temperature indicates the temperature at the altitude. If the pitot tube measures 132,465.16 Pa. what is the true and equivalent air speed of the aircraft?

Given:
h = 6523 m
P_t = 132,465.15 Pa

Step 1: Solve for the temperature at 6523 m

$$T = T0 + ah$$

$$T = 288.2K + (-0.0065 \times 6523m)$$

$$T = 245.8005K$$

Step 2: Solve for the pressure at 6523 m

$$P = P_0 \left[\frac{\mathrm{T}}{T_0}\right]^{5.26}$$

$$P = 101325 \left[\frac{245.8005}{288.2}\right]^{5.26}$$

$$P = 43872.42338\ Pa$$

Step 3: Solve for the Density at 6523m

$$\rho = \rho_0 \left[\frac{\mathrm{T}}{T_0}\right]^{4.26}$$

$$\rho = 1.225 \left[\frac{245.8005}{288.2}\right]^{4.26}$$

$$\rho = 0.6219025176 \ \frac{\mathrm{kg}}{\mathrm{m}^3}$$

Step 4: Solve for True Airspeed

Using the formula, use the actual density at the given altitude.

$$V_{\mathrm{TRUE}} = \sqrt{\frac{2(P_T - P)}{\rho_{\mathrm{actual}}}}$$

$$V_{\mathrm{TRUE}} = \sqrt{\frac{2(132{,}465.15\ \mathrm{Pa} - 43{,}872.42338\ \mathrm{Pa})}{0.6219025176\ \mathrm{kg/m^3}}}$$

$$VTRUE = 533.76796608 m/s$$

Step 5: Solve for the Equivalent Airspeed

Using the formula, use the standard sea level density ρ_0.

$$V_E = \sqrt{\frac{2(P_T - P)}{\rho_{\mathrm{sealevel}}}}$$

$$V_E = \sqrt{\frac{2(132{,}465.15\ \mathrm{Pa} - 43{,}872.42338\ \mathrm{Pa})}{1.225\ \mathrm{kg/m^3}}}$$

$$VE = 380.316896 m/s$$