Problem 4.20

A Pitot tube is mounted in the test section of a low-speed subsonic wind tunnel. The flow in the test section has a velocity, static pressure, and temperature of 150 mi/hr, 1 atm, and 70°F, respectively. Determine the pressure measured by the Pitot tube:

\[ \rho = \frac{p}{RT} = \frac{2116 \text{ lb/ft}^2}{1716 \text{ ft}^2/\text{sec}^2 (530)\text{°R}} = 0.00233 \text{ slugs/ft}^3 \]

For incompressible flow \( \rho = \text{constant} \). Also, \( V = 150 \text{ mi/hr} = 220 \text{ ft/sec} \)

From Bernoulli's equation \( p_o = p + \frac{1}{2} \rho V^2 \)

\[ p_o = 2116 \text{ lb/ft}^2 + \frac{1}{2} (.00233 \text{ slugs/ft}^3)(220 \text{ ft/sec})^2 \]

\[ = 2172 \text{ lb/ft}^2 \]

Note: slug = lb sec²/ft
Problem 4.22

The altimeter on a low-speed airplane reads 2 km. The airspeed indicator reads 50 m/sec. If the outside air temperature is 280°K, determine the true velocity of the airplane (relative to still air at 280°K).

The altimeter measures pressure altitude.

From Appendix A,

\[ p = 7.95 \times 10^4 \text{ N/m}^2 \]

\[ \rho = \frac{p}{RT} = \frac{7.95 \times 10^4}{(287)(280)} = 0.989 \text{ kg/m}^3 \]

The relation between \( V_{\text{true}} \) and \( V_e \) is

\[ \frac{V_{\text{true}}}{V_e} = \sqrt{\rho_s / \rho} \]

Hence,

\[ V_{\text{true}} = 50 \sqrt{(1.225) / 0.989} = 56 \text{ m/sec} \]
Problem 4.23

A Pitot tube is mounted in the test section of a high-speed subsonic wind tunnel. The pressure and temperature of the airflow are 1 atm and 270°K, respectively. The flow velocity is 250 m/sec. Determine the pressure measured by the Pitot tube.

In the test section,

\[ a = \sqrt{\gamma RT} = \sqrt{(1.4)(287)(270)} = 329 \text{ m/sec} \]

\[ M = \frac{V}{a} = \frac{250}{329} = 0.760 \]

\[ \frac{p_o}{p} = \left(1 + \frac{\gamma - 1}{2}M^2\right)^{\frac{\gamma}{\gamma - 1}} = [1 + 0.2 (0.760)^2]^{3.5} = 1.47 \]

\[ p_o = 1.47p = 1.47 (1.01 \times 10^5) = 1.48 \times 10^5 \text{ N/m}^2 \]
Problem 4.26

A high-performance F-16 fighter plane is flying at Mach 0.96 at sea level. Determine the air temperature at the stagnation point at the leading edge of the wing.

At standard sea level,

\[ T = 518.69^\circ R \]

\[ \frac{T_0}{T} = 1 + \frac{\gamma - 1}{2} M^2 = 1 + 0.2 (0.96)^2 = 1.184 \]

\[ T_0 = 1.184T = 1.184 (518.69) \]

\[ T_0 = 614.3^\circ R = 154.3^\circ F \]