<mark>P_{18km} = 7537.31 Pa</mark> ←Answer

Now for the standard atmospheric value of density at 18km altitude,

$$\frac{\rho_{18km}}{\rho_{12km}} = e^{\left(\frac{-g_0}{RT}\right)(h_{18km} - h_{12km})}$$

$$\rho_{18km} = \left(3.1194 \ge 10^{-1} \frac{kg}{m^3}\right) \cdot e^{\overline{\left(287.08 \frac{J}{kg \cdot K}\right)(216.66 \ K)}} (6000 \ m)$$

$$\rho_{18km} = 0.121 \frac{kg}{m^3} \leftarrow \text{Answer}$$

Isothermal region is within the range beginning from h = 11 km above sea level condition to h = 25 km which means temperature is constant. And so,

$$T_{12km} = T_{18km} = 216.66 K \leftarrow \text{ArsereCO}$$

3.2 Consider an airplane flying at some (cal altitude. The outside pressure and temperature 2.65×10^{4} MGn² and 200 K, respectively. What are the pressure and density altitudes?

Given:

$$P_1 = 2.65 \ge 10^4 \frac{N}{m^2}$$
; $T_1 = 200 \text{ K}$; $h_P = ?$; $h_\rho = ?$

Solution:

To solve the pressure altitude by using the given pressure, we need to use and consider the pressure variation at gradient region of 0-11 km.

$$\frac{P_1}{P_0} = \left[\frac{T_1}{T_0}\right]^{5.26}$$

Because $T_1 = T_0 + \lambda h_P$, so

$$\frac{P_1}{P_0} = \left[\frac{T_0 + \lambda h_P}{T_0}\right]^{5.26}$$

By Engr. John Philip Nadela

Chapter 4 Basic Aerodynamics

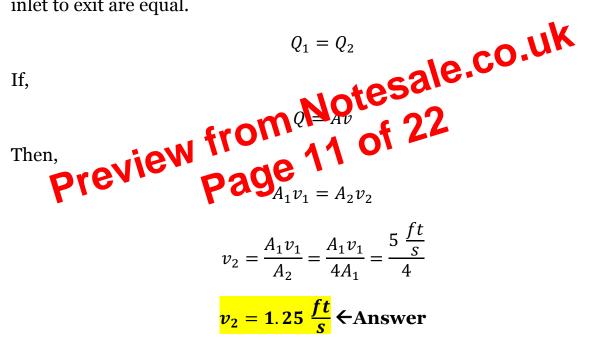
4.1 Consider the incompressible flow of water through a divergent duct. The inlet velocity and area are 5 ft/s and 10 ft², respectively. If the exit area is four times the inlet area, calculate the water flow velocity at the exit.

Given:

$$v_1 = 5 \frac{ft}{s}$$
; $A_1 = 10 ft^2$; $A_2 = 4A_1$; $v_2 = ?$

Solution:

So considering the flow is incompressible which means the density, ρ , is assumed to be constant and has no change, therefore the volume flow rate of inlet to exit are equal.



4.2 In the above problem, calculate the pressure difference between the exit and the inlet. The density of water is $62.4 \text{ lb}_m/\text{ft}^3$.

Given:

$$\rho = 62.4 \ \frac{lb_m}{ft^3}$$
; $P_2 - P_1 = \Delta P = ?$

By Engr. John Philip Nadela

exit, the pressure is 1 atm. Calculate the temperature and density of the flow at the exit. Assume the flow is isentropic and, of course, compressible.

Given:

$$P_R = 10 \text{ atm}$$
; $T_R = 300 \text{ K}$; $P_E = 1 \text{ atm}$; $T_E = ?$; $\rho_E = ?$

Solution:

Let us use the formula for isentropic condition.

$$\begin{bmatrix} T_1 \\ T_3 \end{bmatrix} = \begin{bmatrix} P_1 \\ P_3 \end{bmatrix}^{\frac{k-1}{k}}$$

$$T_3 = \frac{T_1}{\begin{bmatrix} P_1 \\ P_3 \end{bmatrix}^{\frac{k-1}{k}}} = \frac{300 \text{ K}}{\begin{bmatrix} 10 \text{ atm} \\ 1 \text{ atm} \end{bmatrix}^{\frac{1.4-1}{1.4}}}$$

$$T_3 = 155.38 \text{ K} \Leftarrow \text{Ansyon}$$
we we already have the value of pressure and temperature at the

Because we already have the value of pressure and temperature at the exit section, we can use the equation of state to get the density.

$$P_{3} = \frac{P_{3}}{RT_{3}} = \frac{1 \text{ atm} \times \frac{101325 \text{ Pa}}{1 \text{ atm}}}{\left(287.08 \frac{J}{kg \cdot K}\right)(155.38 \text{ K})}$$

$$\rho_{3} = 2.27 \frac{\text{kg}}{\text{m}^{3}} \leftarrow \text{Answer}$$

Chapter 5 Airfoils, Wings, and Other Aerodynamic Shapes

5.3 Consider a rectangular wing mounted in a low-speed subsonic wing tunnel. The wing model completely spans the test section so that the flow "sees" essentially an infinite wing. If the wing has a NACA 23012 airfoil section and a chord of 0.3 m, calculate the lift, drag, and moment about the