C 1080

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Name.....................

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EIGHTH SEMESTER B.TECH. (ENGINEERING) [09 SCHEME] DEGREE EXAMINATION, APRIL 2016

ME 09 802—COMPRESSIBLE FLUID FLOW

Time : Three Hours   Maximum : 70 Marks

Use of gas table is permitted.

Part A

Answer **all** *the questions.*
Each question carries 2 *marks.*

1. A plane travels at a speed of 2400 km/h in an atmosphere of 5°C. Find the mach angle and mention its significance.
2. What is chocked flow in a nozzle ?
3. What is the difference between a nozzle and a diffuser ?
4. The condition of a gas in a combustion chamber at entry are T_1 = 375 K, p_1 = 0.50 bar, c_1 = 70 m/s. The air-fuel ratio is 29 and the calorific value of the fuel is 42 MJ/kg. Calculate final pressure, temperature and velocity of the gas.
5. Shock waves cannot develop in subsonic flow ? Why ?

(5 × 2 = 10 marks)

Part B

Answer any **four** *questions.*

6. An aircraft is flying at an altitude of 11,000 metres, at 800 km/hr. The air is reversibly compressed in an inlet diffuser. The inlet temperature is 216.65 K and pressure is 0.226 bar. If the Mach number at the exit of the diffuser is 0.35, calculate the entry Mach number, Velocity, pressure and temperature of air at the diffuser exit.
7. Derive an expression for the acoustic velocity of a compressible fluid flow in terms of ~~its~~ temperature
8. The condition of a gas in a combustion chamber at entry are T_1 = 375 K, p_1 = 0.50 bar, c_1 = 70 m/s. The air-fuel ratio is 29 and the calorific value of the fuel is 42 MJ/kg. Calculate the initial and final Mach number, final pressure, temperature and velocity of the gas.
9. Describe chocking in Fanno flow along with h-s diagram.

Turn over

10. A convergent-divergent air nozzle has exit to throat area ratio of 3. A normal shock appears at the divergent section where the existing area ratio is 2.2. Find the Mach number, before and after the shock. If the inlet stagnation properties are 500 kPa and 450 K, find the entropy increase across the shock.

11. Describe wave phenomena with types.

(4 × 5 = 20 marks)

Part C

Answer **all** *questions.*

12. (a) Prove that for Isentropic flow :

$$\frac{c^2}{2} + \frac{a^2}{\gamma - 1} = \left(\frac{\gamma + 1}{\gamma - 1}\right)\frac{a^{*2}}{2}.$$

Or

(b) Air (Cp = 1.05 kJ/kgK, γ = 1.38) at $p_1 = 3 \times 10^5$ N/m² and T_1 = 500 K flows with a velocity of 200 m/s in a 30 cm. diameter duct available. Calculate (i) Mass flow rate ; (ii) Stagnation temperature ; (iii) Mach number ; and (iv) Stagnation pressure values.

13. (a) A supersonic nozzles air expands from p_0 = 24 bar and T_0 = 1000K to an exit pressure of 4.3 bar. If the exit area of the nozzle is 110 cm², calculate the following :—

(i) Throat area ; (ii) Pressure and temperature at the throat ; (iii) Temperature at exit ; (iv) Mass flow rate ; and (v) Exit velocity as fraction of the maximum attainable velocity.

Or

(b) Derive area ratio as a function of Mach number for Isentropic flow :

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

14. (a) The condition of a gas in a combustion chamber at entry are T_1 = 375 K, p_1 = 0.50 bar, c_1 = 70 m/s. The air-fuel ratio is 29 and the calorific value of the fuel is 42 MJ/kg. Calculate :

(i) The initial and final Mach number.

(ii) Final pressure, temperature and velocity of the gas.

(iii) Percentage of stagnation pressure loss.

(iv) Maximum stagnation temperature.

Or

(b) Show that the ratio of pressure between any two section in terms of their Mach number in a Fanno flow

$$\frac{p_2}{p_1} = \frac{M_1}{M_2}\left[\frac{1+\frac{\gamma-1}{2}M_1 2}{1+\frac{\gamma-1}{2}M_2 2}\right]^{\frac{1}{2}}.$$

15. (a) A convergent-divergent nozzle is designed to expand air from a reservoir in which the pressure is 700 kPa and temperature is 5°C and the nozzle inlet Mach number is 0.2. The nozzle throat area is 46 cm^2 and the exit area is 230 cm^2. A normal shock appears at a section where the area is 175 cm^2. Find the exit pressure and temperature. Also find the increase in entropy across the shock.

Or

(b) Derive the temperature ratio across the shock.

$$\frac{T_y}{T_x} = \frac{\left[\frac{2\gamma}{\gamma-1}M_x^{\ 2}-1\right]\left[1+\frac{\gamma-1}{2}M_x^{\ 2}\right]}{\frac{M_x^{\ 2}}{2(\gamma-1)}(\gamma+1)^2}.$$

(4 × 10 = 40 marks)