Seat No.: _____ Enrolment No.______

GUJARAT TECHNOLOGICAL UNIVERSITY

BE - SEMESTER–VIII • EXAMINATION – SUMMER • 2015

Subject code: 181906 **Date:05/05/2015**

Subject Name: Gas Dynamics

Time: 10.30AM-01.00PM **Total Marks: 70**

Instructions:

1. **Attempt all questions.**
2. **Make suitable assumptions wherever necessary.**
3. **Figures to the right indicate full marks.**
4. **Use of Gas Table is permitted.**

Q.1 **(a)** Explain the flow through convergent and Convergent - divergent nozzle with proper pressure v/s distance diagram. **07**

(b) Derive the following equation and draw the shape of nozzle and diffuser for subsonic, sonic and supersonic flow. $\frac{dA}{A} = \frac{dp}{\rho c^2}$ $(1-M^2)$ **07**

Q.2 **(a)** Explain how does a shock wave develop in the diverging section of a supersonic nozzle? How does this wave move towards the exit? **07**

(b) Air (γ=1.4, R=287.43 J/kg K) enters a straight axisymetric duct at 300 K, 3.45 bar and 150 m/s and leaves it at 277 K, 2.058 bar and 260 m/s. The area of cross section at entry is 500 cm^2. Assuming adiabatic flow determine (i) stagnation temperature (ii) maximum velocity (iii) mass flow rate (iv) area of cross section at exit. **07**

OR

(b) Derive equivalent Bernoulli's equation for isentropic compressible flow. **07**

$$\frac{\gamma}{\gamma-1}\frac{P_0}{\rho_0}\left(\frac{P}{P_0}\right)^{\frac{\gamma-1}{\gamma}} + \frac{1}{2}c^2 = \frac{\gamma}{\gamma-1}\frac{P_0}{\rho_0}$$

Q.3 **(a)** Show that the upper and lower branches of a Fanno curve represent subsonic and supersonic flows respectively. Prove that at the maximum entropy point Mach number is unity and all processes approach this point. **07**

(b) A long pipe of 25.4 mm diameter has a mean coefficient of friction of 0.003. Air enters the pipe at a Mach number of 2.5, stagnation temperature 310 K and static pressure 0.507 bar. Determine for a section at which the Mach number reaches 1.2, (i) static pressure and temperature (ii) stagnation pressure and temperature (iii) velocity of air (iv) distance of this section from the inlet (v) mass flow rate of air. **07**

OR

Q.3 **(a)** Derive an equation describing a Fanno curve. Show three Fanno curves on temperature v/s entropy coordinates at three mass flow densities. **07**

(b) Explain the direction of isothermal flow process with friction with neat T-s diagram **07**

Q.4 **(a)** What is Rayleigh flow? Give two practical examples. Under what conditions the assumption of Rayleigh flow is not valid in a heat **07**

exchanger?

(b) The conditions of a gas in a combustor at entry are: p_1=0.343 bar, T_1=310 K, c_1=60 m/s; Determine the Mach number, pressure, temperature and velocity at the exit, if the increase in stagnation enthalpy of the gas between entry and exit is 1172.5 kJ/kg. Take cp=1.005 kJ/kg K, γ=1.4. **07**

OR

Q.4 (a) Prove that Mach numbers at the maximum enthalpy and maximum entropy points on the Rayleigh line are $1/\sqrt{\gamma}$ and 1.0 respectively. Show the h=constant and s=constant lines at these points on the Rayleigh line on the h-s and p-v planes. **07**

Q.4 (b) The stagnation temperature of air in a combustion chamber is increased to 3.5 times its initial value. If the air at entry is at 5 bar, 105^0C and a Mach number of 0.25. Determine (a) the Mach number, pressure and temperature at the exit, (b) stagnation pressure loss and (c) the heat supplied per kg of air. **07**

Q.5 (a) Starting from the energy equation for flow through a normal shock, obtain the following relations: $C_x C_y = a^{*2}$ and $M_x^{\ *} M_y^{\ *} = 1$ **07**

(b) A jet of air at 275 K and 0.69 bar has an initial Mach number of 2.0. If it passes through a normal shock wave. Determine (i) Mach number (ii) pressure (iii) temperature (iv) density (v) speed of sound and (vi) jet velocity at downstream of the shock. **07**

OR

Q.5 (a) Derive the following relations for flow through a normal shock: **07**

(a) $M_y^{\ 2} = \dfrac{\left[\dfrac{2}{\gamma-1}\right] + M_x^{\ 2}}{\left[\dfrac{2\gamma}{\gamma-1}\right] M_x^{\ 2} - 1}$ (b) $\dfrac{P_y}{P_x} = \dfrac{2\gamma}{\gamma+1} M_x^{\ 2} - \dfrac{\gamma-1}{\gamma+1}$

(b) A Mach -2 aircraft engine employs a subsonic inlet diffuser of area ratio 3. A normal shock is formed just upstream of the diffuser inlet. The free stream conditions upstream of the diffuser are: p= 0.10 bar, T= 300 K. Determine (a) Mach number, pressure and temperature at the diffuser exit. (b) Diffuser efficiency including the shock. Assume isentropic flow in the diffuser downstream of the shock. **07**

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