Reg. No. | | | | | | | | | | | |

Question Paper Code : 80669

B.E./B.Tech. DEGREE EXAMINATION, NOVEMBER/DECEMBER 2016.

Sixth Semester

Mechanical Engineering

ME 6604 — GAS DYNAMICS AND JT PROPULSION

(Regulations 2013)

Time : Three hours Maximum : 100 marks

Answer ALL questions.

PART A — ($10 \times 2 = 20$ marks)

1. A gas flows through a restricted passage with a speed of 850 m/s. Its local temperature is 1650 K; its specific heat ratio k and gas constant R are 1.25 and 250 J/kg K respectively. Calculate the local sonic velocity and Mach number.

2. Define Crocco number.

3. Define critical condition in Fanno flow.

4. Write down the expression for the pressure ratio of two sections in terms of Mach number in Rayleigh flow?

5. Mention the useful applications of shock wave.

6. Define strong and weak wave.

7. What is after burning in turbojet engines?

8. Differentiate between pressure thrust and momentum thrust.

9. Define escape velocity.

10. Name any two solid propellant fuels and oxidizers.

PART B — (5 × 16 = 80 marks)

11. (a) (i) Given speeds and Mach numbers, assuming air is a perfect gas, determine the corresponding local temperature (take 1 mi/hr = 0.447 m/s) for the following: (1) A Boeing 747-400 at a cruise speed of 910 km/hr; M = 0.85. (2) Concorde at a cruise speed of 1,320 mi/hr; M = 2.0 (3) The fastest airplane, the Lockheed SR-71 Blackbird, flying at 2,200 mi/hr; M = 3.3 (4) The fastest car, the Thrust SSC, averaged 760.035 mi/hr; M = 0.97. (8)

(ii) Helium flows at Mach 0.5 in a channel with cross-sectional area of 0.16 m^2. The stagnation pressure of the flow is 1 MPa, and stagnation temperature is 1,000 K. Calculate the mass flow rate through the channel, with $\gamma = 5/3$. (8)

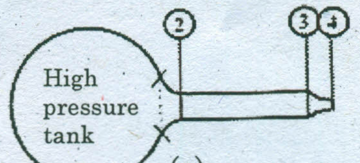
Or

(b) A supersonic diffuser, diffuses air in an isentropic flow from a Mach number of 3 to a Mach number of 1.5, the static conditions of air at inlet are 70 kPa and –7°C. If the mass flow rate of air is 125 kg/s, determine (i) Stagnation conditions, (ii) Area at throat and exit, (iii) Static conditions of air at exit. (16)

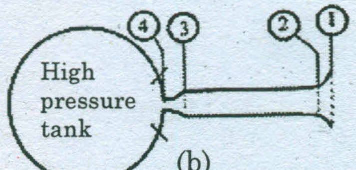
12. (a) Air ($\gamma = 1.4$) flows into a constant-area insulated duct with a Mach number of 0.20. For a duct diameter of 1 cm and friction coefficient of 0.02, determine the duct length required to reach Mach 0.60. Determine the length required to attain Mach 1. Finally, if an additional 75 cm is added to the duct length needed to reach Mach 1, while the initial stagnation conditions are maintained, determine the reduction in flow rate that would occur. (16)

Or

(b) (i) In which configuration of Figure (a) or (b), will the high-pressure tank empty faster? Explain. (6)



(a)



(b)

(ii) The stagnation temperature of air is raised from 85°C to 376°C in a heat exchanger. If the inlet Mach number is 0.4, determine the final Mach number and percentage drop in pressure. (10)

13. (a) State and prove Prandtl – Meyer relation for a normal shock. (16)

Or

(b) A gas at a pressure of 340 mbar, temperature of 355 K and entry Mach number of 1.4 is expanded isentropically to 140 mbar. Calculate the following (i) deflection angle, (ii) final Mach number, (iii) Final temperature of the gas. Take $\gamma = 1.3$. (16)

14. (a) Differentiate turbojet and turboprop propulsion engines with suitable diagrams. (16)

Or

(b) A ramjet engine operates at M = 1.2 at an altitude of 6500 m. The diameter of inlet diffuser at entry is 50 cm and the stagnation temperature at the nozzle entry is 1500 K. The calorific value of the fuel used is 40 MJ/kg. The properties of the combustion gases are same those of air ($\gamma = 1.4$, R = 287 J/kg K). The velocity of the air at the diffuser exit is negligible, calculate: (i) the efficiency of the ideal cycle, (ii) Flight speed, (iii) Air flow rate, (iv) Diffuser pressure ratio, (v) Fuel air ratio, (vi) Nozzle jet Mach number. The efficiencies of the diffuser = 0.9, combustor = 0.98 and the nozzle = 0.96 (16)

15. (a) Describe the important properties of liquid and solid propellants desired for rocket propulsion. (16)

Or

(b) Calculate the thrust, specific impulse, propulsive efficiency, thermal and overall efficiencies of a rocket engine from the following data: Effective jet velocity = 1250 m/s, Flight to jet speed ratio = 0.8, oxidizer flow rate = 3.5 kg/s, fuel flow rate = 1 kg/s, heat of reaction of exhaust gases = 2,500 kJ/kg. (16)

———————