Roll. No | | | | | | | | | | |

52

B.E/B.Tech (Full-Time) DEGREE END SEMESTER EXAMINATIONS, NOV/DEC 2011

MECHANICAL ENGINEERING BRANCH

SIXTH SEMESTER

ME 9026 – GAS DYNAMICS & JET PROPULSION

(REGULATIONS 2008)

Time : 3 hr Max Mark : 100

Instructions: 1. State clearly any assumption made with justification

(Use of Approved Gas Tables data book is allowed)

Answer ALL questions

Part A – (10 × 2 = 20 mark)

1. What are the differences between sound wave and shock wave?
2. Why CD nozzles are sometimes produce 'venturi effect'?
3. What are the governing equations used to describe the Fanno flow?
4. How the fluid total temperature and pressure varies during Rayleigh flow process?
5. State the reason for impossibility of shock in subsonic flow.
6. How oblique shock differs from normal shock?
7. What is hybrid aircraft propulsion system?
8. How do you define thermal and overall efficiency of aircraft propulsion system?
9. Why the liquid propellant rockets are more complicated than solid propellant?
10. Classify the rocket engines based on sources of energy employed?

Part B – (5 × 16 = 80 mark)

11.a) i Air flows through a C-D nozzle from a reservoir where stagnation temperature is known to be 60°C. At some section x in the diverging section, a shock occurs. The location of the shock is such that the static pressure measured at the throat is 0.8 times the total pressure measured after the shock. If the flow is isentropic except across the shock, determine (i) the area ratio A_x/A_{throat} (ii) the air velocity behind the shock. **(10)**

ii Explain the phenomena of normal shock with the help of Fanno line and Rayleigh line on the same h-s plot **(6)**

12.a) A source of disturbance travels in air at subsonic, sonic and supersonic velocities. Sketch and explain the propagation of disturbance in each of the cases.

OR

b) Air is discharged from a reservoir at 1 MPa and 500 K through a nozzle to an exit pressure of 0.09 MPa. If the flow rate through the nozzle is 3600 kg/h, determine for isentropic flow (i) throat area, pressure and velocity (ii) exit area and Mach number

13.a) Air enters a constant area pipe with velocity 150 m/s, temperature 60°C and pressure 0.5 MN / m^2. If 180 kJ / kg of heat is added to the pipe find (i) the final pressure (ii) final Mach number (iii) change in stagnation pressure and (iv) change in entropy.

OR

b) Air is flowing in an insulated duct with the entry Mach number 0.25 and friction factor 0.001. The diameter of the duct is 15 cm. (i) What length of the pipe would give 10% of stagnation pressure loss? (ii) Also find % of stagnation pressure loss from inlet to exit Mach number 0.8. (iii) What is the maximum length required to reach the chocking condition.

14 a) A turbojet engine flying at a speed of 900 km/h is fitted with a CD nozzle having an exit diameter 500 mm and area ratio 2. The exit from the turbine of the engine is 300 kPa and 500 K. Find the thrust developed by the engine if the atmosphere pressure at altitude is 20 kPa. Assume properties of the exhaust gas to be same as that of air. Fuel air ratio for the engine is 0.02167.

OR

b) A ramjet engine is designed to operate at a Mach number of 2 at 8 km altitude. A divergent conical inlet diffuser is employed and is designed such that a normal shock is formed at the entrance of the diffuser and the Mach number at the exit of the diffuser is 0.2. Heat added in the combustor is such that the total temperature of the gases at inlet to the choked nozzle is 2000 K. Find the area ratio required for the diffuser. Also find out the heat to be added in the combustor per kg of gas flow through the nozzle and the Mach number at inlet to the nozzle. Assume that the internal flow is frictionless.

15 a) A rocket engine has the following performance details: velocity of jet 1400 m/s; flight to jet speed ratio 0.8; oxidizer flow rate 4.0 kg/s; fuel flow rate 1.0 kg/s and heat of reaction per kg of exhaust gas 2500 kJ/kg. Calculate the thrust, specific impulse, propulsive efficiency, thermal efficiency and overall efficiency of the rocket engine.

OR

b) A liquid propellant rocket engine operating with a chamber pressure of 20 bar develops a thrust of 7000 N while operating at an altitude where the ambient pressure is 10 mbar. The nozzle exit diameter is 225 mm and the exit pressure of gases is 50 mbar. The propellant flow rate is 3 kg/s. Calculate the (i) jet velocity (ii) specific impulse (iii) combustion temperature (iv) thrust coefficient and (v) propellant mass flow coefficient. Take specific heat ratio = 1.4 and gas constant = 287 J/kg. K