Question Paper Code : 71030

B.E./B.Tech. DEGREE EXAMINATION, APRIL/MAY 2015.

Fifth Semester

Aeronautical Engineering

AE 2304/AE 1351/AE 71/080180029/10122 AE 504 – PROPULSION – II

(Regulation 2008/2010)

Time : Three hours Maximum : 100 marks

(Gas tables may be permitted)

Answer ALL questions.

PART A — ($10 \times 2 = 20$ marks)

1. Explain why increasing the flight speed of an ideal ramjet engine increases the thermal efficiency.

2. List out the different methods of turbine blade cooling.

3. How does the propellant grain design affect the performance of the rocket propulsion system?

4. What Kind of Liquid Propulsion Systems do GEO Satellites Use?

5. Define Specific Impulse.

6. Name any two properties of hybrid propellant.

7. Describe briefly three important applications of rocket propulsion.

8. Explain briefly why turbine blades in a multi-stage machine are longer in the back stages than in the front stages.

9. For a given thrust and specific impulse, an ion engine is physically larger than a Hall Thruster. Explain why.

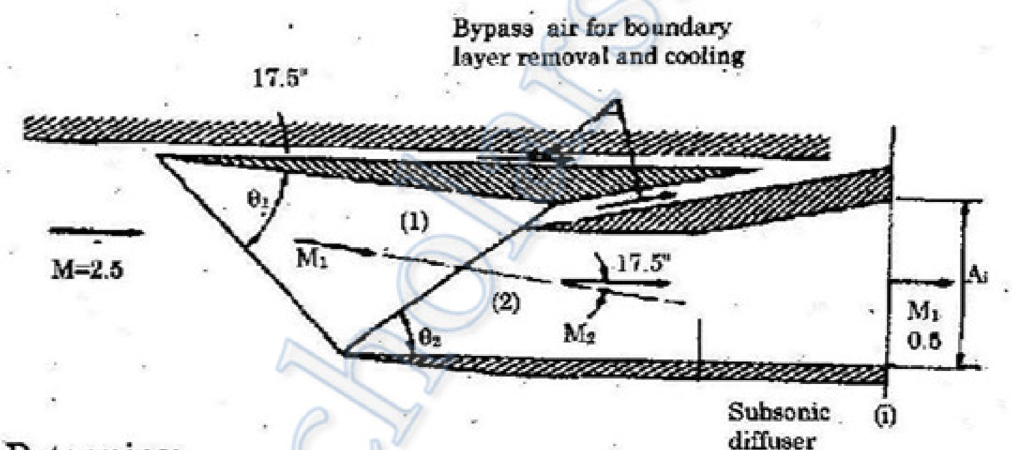
10. A rocket engine produces a thrust of 1000 kN at sea level with a propellant flow rate of 400 kg/s. Calculate the specific impulse.

PART B — (5 × 16 = 80 marks)

11. (a) A ramjet engine is to fly at a Mach number of 4 in an atmosphere whose temperature is 223 K. At entrance to the burner, the Mach number of the flow is 0.3. Combustion in the burner (whose cross-sectional area is constant) may be represented approximately as heating of a perfect gas with constant specific heat ratio. At the exit from the burner the temperature of the gas is 2462 K. Neglecting frictional effects in the burner and considering the flow to be one-dimensional throughout, estimate the Mach number of the gas leaving the burner. Determine also the stagnation pressure loss due to heating (i.e., calculate the ratio of outlet and inlet stagnation pressures).

Or

(b) A new supersonic passenger aircraft is being designed for flight Mach number 2.5 at an altitude where the ambient pressure and temperature are 9 kPa and 220 K, respectively. The engine inlet configuration shown in the figure below allows for double oblique shock deceleration followed by a zone of subsonic deceleration. The Mach number is 0.5 at the engine inlet plane. Losses in the subsonic diffuser are neglected.



Determine:

(i) The Mach numbers M_1 and M_2 in the zones (1) and (2) shown on the drawing.

(ii) The wave angle θ_1 and θ_2 also shown on the drawing.

(iii) The overall stagnation pressure ratio P_{0i}/P_{0a}

(iv) The overall static pressure ratio P_i/P_a

(v) The velocity ratio c_i/c_2 for the subsonic diffuser, and

(vi) The cross-section area $A_i(m^2)$ at the engine inlet plane if the engine mass flow rate is 500 kg/s.

12. (a) An axial turbine stage has the hub-tip ratio of $\zeta = 0.6$. At the mean radius it has: work ratio $\Delta h_0 / U^2 = 1.2$, axial velocity ratio $c_z/U = 0.6$, and degree of reaction R=0.5. Two swirl distributions are being considered for the flow:

One is free vortex and the other is defined by $rc_\theta = a(r + r_m)$ in which r_m is the mid radius. If the stage work and stagnant conditions are radially uniform in the turbine rotor, determine the hub and tip velocity triangles for these two cases and identify the significant differences.

Or

(b) In a preliminary design study of an axial compressor stage, the assumption of axial velocity ratio proportional to radius, that is, $c_z/U \, \alpha \, (c_z/U)_t$ is being explored. The subscript t denotes the tip radius. At the tip radius the tangential velocity ratio $c_{\theta 1}/U$ and the work ratio $(\Delta c_\theta/U)$, are specified, and the inlet stagnation pressure and temperature, P_0 and T_0 are known. The compressor work per unit mass is to be independent of radius. In terms of the parameter specified above, determine the distribution with radius of the tangential velocity ratio $c_{\theta 1}/U$ at the inlet to the rotor.

13. (a) The hot gas supply to a propulsion nozzle is at a pressure of 55 kPa and a temperature of 800 K. The ambient pressure is 15 kPa. If the ratio of specific heats is 1.33, approximately constant, and the molecular weight of the mixture is 30, determine the ratio of gross thrusts for two cases:

(i) The nozzle is converging-diverging with the area ratio for correct expansion to ambient pressure.

(ii) The nozzle is converging only.

What is the nozzle-exit-to-throat area ratio for case (i)? The gross thrust is the contribution to engine thrust made by the exhaust nozzle only.

Or

(b) Explain the different types of propulsion systems which are used in space.

14. (a) (i) Explain briefly the following terms in solid propellant rocket

(1) Linear burning rate (2)

(2) Combustion rate (2)

(3) Propulsion area ratio (2)

(4) Equilibrium combustion pressure (2)

(ii) Draw a neat sketch of a turbo pump feed system employed in rockets. (8)

Or

(b) (i) How is regressive, neutral and progressive burning of the solid propellant grain achieved? Explain with the aid of diagram. (8)

(ii) List out the various selection criteria for solid and liquid rocket propellants. (8)

15. (a) Find the ratio of the velocities of two vehicles, one powered by a liquid-chemical rocket and the other by a solid-chemical one, when they are used for acceleration of a 10,000-kg payload in a zero-gravity field. Both vehicles have a total initial mass of 510,000 kg. The liquid-propellant rocket has 60% greater specific impulse and 30% greater mass of empty vehicle (without propellant and payload), and the solid-propellant rocket has a structural coefficient of $\varepsilon = 0.080$.

Or

(b) Using a propellant of molecular weight 15 and flame temperature 3300 K, determine the rocket throat and exhaust areas required for a thrust of 500 kN and an ideal specific impulse of 300 sec. The ambient pressure is 0.1 MPa, and the specific-heat ratio of the propellant is 1.4. How much thrust would this rocket develop if the ambient pressure were changed to 0.03 Mpa? How much thrust would be developed by a rocket designed to expand to 0.03 MPa if it had the same stagnation conditions, throat area and propellant?