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B.M.S. College of Engineering, Bengaluru-560019

Autonomous Institute Affiliated to VTU

August 2024 Supplementary Examinations

Programme: B.E. **Semester: V**
Branch: Aerospace Engineering **Duration: 3 hrs.**
Course Code: 20AE5DCAAD **Max Marks: 100**
Course: Advanced Aerodynamics

Instructions: 1. Answer any FIVE full questions, choosing one full question from each unit.
2. Missing data, if any, may be suitably assumed.
3. Use of Gas Tables is permitted.

UNIT - I

1 a) Derive the Area-Velocity and Area-Pressure relation and explain how a convergent and divergent passages behave as a nozzle and diffuser for a subsonic Mach entry and vice versa for a supersonic Mach entry. **10**

b) An air nozzle is to be designed for an exit Mach number of 3.5. The stagnation pressure and temperature for the isentropic flow are 800 kPa and 240ºC. Estimate the pressure, temperature, velocity and area at throat and exit for a mass flow rate of 3.5 kg/s. **10**

OR

2 a) Describe the flow through converging and converging-diverging nozzle with the help of a pressure-distance diagram and Mach number-distance diagram (for CD-nozzles only). Compare the nozzle choking in both converging and converging-diverging nozzles with the help of a mass flow rate versus P_b/P_o diagram. **10**

b) Define stagnation enthalpy, stagnation pressure and stagnation temperature. **6**

c) A tank has 25 kg of an ideal gas at a temperature of 18°C.The volume of the tank is 5 m^3 and the molecular weight of the gas is 28. R = 8314 J/kg–mole–k. What is the pressure inside the tank? **4**

UNIT – II

3 a) Derive the Prandtl’s relation for a normal shock wave. **10**

b) An airstream (γ = 1.4 and R = 287 J/kg-K) with a velocity of 500 m/s a static pressure of 50 kPa and a static temperature of 250 K undergo a normal shock. Determine the air velocity and the static and stagnation conditions after the wave. **10**

OR

4 a) Explain how velocity and Mach number varies along the shock tube after bursting the diaphragm. **5**

b) Air approaches a symmetrical wedge (δ = 15°) at a Mach number of 2.0. Determine for the strong and weak shock waves (a) wave angle, (b) pressure ratio, (c) density ratio, (d) temperature ratio and (e) downstream Mach number. **15**

UNIT - III

5 a) A jet of air at a Mach number of 2.5 is deflected inwards of a corner of a curved wall, the wave angle at the corner is 60°. Determine the deflection angle of the wall, pressure and temperature ratios and the final Mach number. **10**

b) A uniform supersonic flow at Mach $M_1 = 2.0$, $P_1 = 0.8$ bar and temperature 270 K expands through two convex corners of 10^0 each as shown in the figure 1. Determine the downstream Mach number M_3, pressure p_2, temperature T_2 and the angle of the fan. **10**

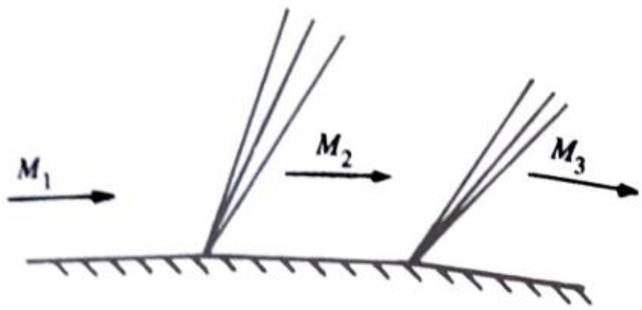

Figure 1.

UNIT - IV

6 a) Air is flowing in an insulated duct with friction co-efficient 0.002. At inlet velocity of air 130 m/s, pressure 250 kPa and temperature 400 K. The diameter of the duct is 16 cm. Determine (i) the length of the pipe required so as to give 20 % drop in stagnation pressure and (ii) properties of air at section from 3.5 m from inlet and (iii) Maximum length of the pipe **10**

b) The Mach number at the exit of the of a combustion chamber is 0.9. The ratio of stagnation temperatures at the exit and entry is 3.74. If the pressure and the temperature of the gas at exit are 2.5 bar and 1000°C respectively. Determine (i) Mach number, pressure and temperature of the gas at entry, (ii) the heat supplied per kg of the gas and (iii) the maximum heat that can be supplied. **10**

UNIT -V

7 a) Derive the linearised potential equation for a subsonic, transonic and supersonic flow using small perturbation theory. **10**

b) A two-dimensional airfoil has a thickness ratio (maximum thickness to the chord) of 0.04 and a camber ratio of 0.015. When tested in a low-speed wind tunnel (incompressible flow $M_\infty = 0$) at an angle of attack of 3°, the lift co-efficient C_L is measured to be 0.6. It is desired to determine the performance of a similar airfoil at $M_\infty = 0.5$ and Prandtl Glaueret rule, Determine the geometrical characteristics and the lift-coefficient of the related airfoil in compressible flow at the given M_∞ using the Gothert's rule for the same airfoil. **10**

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