5E3177

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B.Tech. (Sem.V) (Main/Back) Exam Dec. 2012

Mechanical Engineering

5ME3 Fundamentals of Aerodynamics

[Time : 3 Hours] [Total Marks : 80]

[Min. Passing Marks : 24]

Instructions to Candidates :

*Attempt any **five questions**, selecting **one question** from **each unit**. All questions carry equal marks. Schematic diagrams must be shown wherever necessary. Any data you feel missing suitably be assumed and stated clearly. Units of quantities used/calculated must be stated clearly.*

Unit - I

1. (a) What are the sources of aerodynamics forces and moments over the body surfaces? Explain in detail. **(8)**

   (b) Explain the centre of pressure for an aerodynamic body. **(8)**

OR

1. (a) Starting with the definition of circulation, derive Kelvin's circulation theorem. **(8)**

   (b) What is an aerofoil? Explain various terms associated with aerofoil nomenclature indicating them on an aerofoil sketch. **(8)**

Unit - II

2. (a) A jet plane which weighs 30,000 N and has a wing area of $20\ m^2$ flies at a velocity of 250 km/h. When the engine velocity 750 W, 65% of the power is used to over come the drag resistance of the wing. Calculate the co-efficient of lift and drag for the wing. Take density of air equal to $1.21\ kg/m^3$. (8)

(b) Describe the important nomenclature of a turbine cascade with a neat sketch. (8)

OR

2. (a) Derive the expression for cascade lift and drag. (8)

(b) Differentiate between symmetrical and non-symmetrical aerofoil. (8)

Unit - III

3. (a) Derive the following relations for isentropic flow.

(i) $$\frac{dP}{\rho} = \left[\frac{M^2}{1-M^2}\right]\frac{dA}{A}$$

(ii) $$\frac{dV}{V} = \left[\frac{1}{1-M^2}\right]\frac{dA}{A}$$

where all the notations have their usual meanings. (4×2=8)

(b) Sketch the variation of area, velocity and pressure for isentropic flow through subsonic and supersonic nozzle. (8)

OR

3. (a) Show that the mass flow through a choked nozzle is

$$m^* = \frac{P_0 A^*}{\sqrt{T_0}}\sqrt{\frac{\gamma}{R}\left(\frac{2}{\gamma+1}\right)^{\frac{(\gamma+1)}{(\gamma-1)}}}$$

where all the notations have their usual meanings. (8)

(b) In an airflow where the local mach number, static pressure and static temperature are 3.5, 0.3 atm and 180 k respectively. Calculate the local values of P_0, T_0, T^*, a^* and M^* at this point. (8)

Unit - IV

4. Explain the Fanno line and Rayleigh line in detail with h-s plane. (16)

OR

4. (a) A constant area combustion chamber is supplied air at 1.2 bar and mach number 1.5. At the exit the mach number changes to 3. Determine exit pressure and direction of heat transfer.
[take $\gamma = 1.4$] (8)

(b) In a Rayleigh flow, if value of mach number at exit is 0.93, stagnation temperature is 300° C and at inlet stagnation temperature is 100° C, find the mach number at inlet and ratio of exit pressure and inlet pressure.
[take $\gamma = 1.4$] (8)

Unit - V

5. (a) Derive the following relationship for a normal shock wave :

$$C_x C_y = (a^*)^2$$

where, C_x = Velocity before shock

C_y = Velocity after shock

a^* = Critical velocity of sound. (8)

(b) Define the equation for Rankine-Hugoniot relations for normal shock. (8)

OR

5. (a) Define the equation for static pressure ratio across two shock. (8)

(b) Consider a normal shock wave in a supersonic air stream where the pressure upstream of the shock is 1 atm. Calculate the loss of total pressure across the shock wave when the upstream mach number is $M_1 = 2$. (8)