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MANIPAL INSTITUTE OF TECHNOLOGY
MANIPAL
(A constituent unit of MAHE, Manipal)

V SEMESTER B.TECH. (AERONAUTICAL ENGINEERING)
END SEMESTER EXAMINATIONS, NOV/DEC 2018

SUBJECT: GAS DYNAMICS [AAE 3102]

REVISED CREDIT SYSTEM
(21/11/2018)

Time: 3 Hours MAX. MARKS: 50

Instructions to Candidates:

- Answer **ALL** the questions.
- Missing data may be suitable assumed.
- Gas tables will be provided by the department

1A. What is compressibility effect and how it's affecting the flow properties when speed changes from low subsonic to high subsonic? Also calculate the isothermal compressibility for air at a pressure of 0.8atm. **(02)**

1B. Derive and prove how incompressible pressure coefficient is related to compressible pressure coefficient through linearized theory. Also explain why this theory is not being applied to all type of flows. **(04)**

1C. Consider a supersonic flow passing through a constant area duct with diameter of 0.5m. The inlet conditions of the flow are M_1=2.2, P_1=1.2atm, T_1=292K. At the starting of the duct heat being added by amount of q=10^4 J/Kg and followed by rough surfaces for the length of 0.5m with friction coefficient f=0.005. Calculate the downstream properties of the flow at the end of the duct. **(04)**

2A. Draw the schematic diagram and Pressure-Deflection diagram for intersection of opposite families with running curve deflection angle is higher than left running curve deflection angle. **(02)**

2B. Derive and prove that Mach number at any location in the duct is a function of the ratio of the local duct area to the sonic throat area. Also prove that for supersonic wind tunnel diffuser throat area is always bigger than inlet throat area. **(04)**

2C. Consider a supersonic nozzle with a Pitot tube mounted at the exit. The reservoir pressure and temperature are 10atm and 500K, respectively. The pressure measured by the Pitot tube is 0.6172atm. The throat area is 0.3m^2. Calculate the exit Mach number (M_e), Exit Area (A_e), Exit pressure (P_e), Exit temperature (T_e) and Mass flow through the nozzle. **(04)**

3A. Explain the reason for occurrence of Mach reflection and what is meant by "choking" of the flow? Write an example **(02)**

3B. Derive and prove the following relation for an expansion wave and write down its procedures to calculate the downstream properties of the flow. **(04)**

$$\theta=\nu(M_2)-\nu(M_1)$$

3C. Consider the following Figure-1 and a supersonic flow entering into region 1 with M_1=2.4, P_1=1.3atm and T_1=300K. (deflection angle is 15^0 in all cases) Calculate the following downstream properties of the flow: **(04)**

a) Mach number at region 5
b) Pressure and temperature at region 4
c) Density of flow at region 3
d) Total pressure at region 3

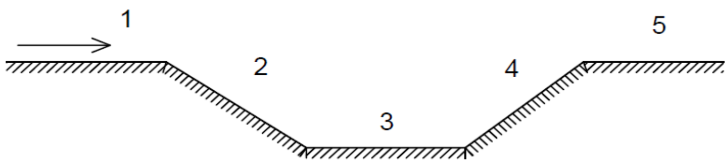

Figure-1

4A. Write down the procedures to determine downstream properties of the flow from the Velocity Potential equation **(02)**

4B. Derive and prove that Mach number behind the normal shock is always a subsonic flow. Also derive how local downstream pressure and Mach number related to inflow local pressure and Mach number. **(04)**

4C. A supersonic flow with M_1=3, P_1=1.1atm and T_1=300K passing through a compression corner in Figure-2. The incident shock is reflected at point B and creating a reflected shock which propagates downstream. **(04)**
Calculate Mach number, pressure, temperature and reflected shock angle behind the reflected shock.

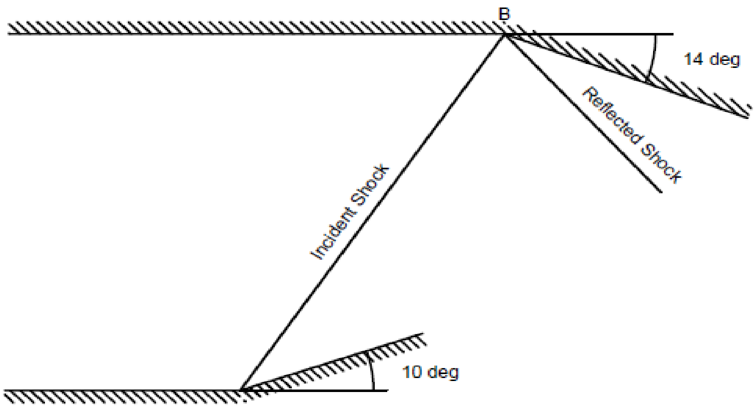

Figure-2

5A. What is the difference between normal shock and oblique shock? How these shocks affecting the flow properties? **(02)**

5B. Derive and prove that deflection angle is unique function of Mach number and wave angle. **(04)**

5C. Consider an aircraft with plan form area $20m^2$ in steady level flight at Mach 2 at altitude of 11km(ρ=0.365kg/m^3, T_∞=217K). The weight of the aircraft is 9500kgf. Assume that all the lift of the airplane comes from the lift on the wings also assuming the lift coefficient of the wing is the same as the lift coefficient for the airfoil section. The chord length of the airfoil is 2.4m which almost equal to the mean chord length of the wing. calculate the following: **(04)**

a) Calculate the angle of attack with the help of linearized equations
b) If its fully turbulent flow calculate the coefficient of friction and wave drag
c) At what angle of attack C_f is equal to C_D