Bengal Engineering and Science University, Shibpur
BE (Aero) Part III 6th Semester Final Examination, 2013

High Speed Aerodynamics I
(AE 601)

Time : 3 hours | Full Marks : 70

Answer any five of the following.
The questions are of equal value.
Compressible flow table can be obtained on demand
And is to be returned after use

1. a) Show that $p_0A^* =$ constant across a shock wave inside a nozzle.

   b) Consider a convergent-divergent nozzle with an exit-to-throat area ratio of 3. The inlet reservoir is 1 atm and the exit static pressure is 0.5 atm. For this pressure ratio a normal shock will stand inside the divergent portion of the nozzle. Calculate the location of the shock wave.

2. Consider a rocket engine burning hydrogen and oxygen; the combustion chamber temperature and pressure are 3517 K and 25 atm respectively. The molecular weight (M) of the chemically reacting gas in the combustion chamber is 16 and specific gas constant and universal gas constant are 519.6 J/(kg.K)and 8314 J/(kg.mol.K) and γ = 1.22. The pressure at the exit of the convergent-divergent rocket nozzle is $1.174 X 10^{-2}$ atm. The area of the throat is 0.4 m^2. Assuming calorically perfect gas and isentropic flow, calculate (a) exit Mach number, (b) the exit velocity (c) the mass flow through the nozzle and,(d) the area of the exit.

3. Consider an infinitely thin flat plate at an angle of attack of 20^0 in a Mach 3 free stream. Calculate the magnitude of the flow direction angle Φ downstream the trailing edge.

4. Consider a 15^0 half-angle diamond wedge airfoil is in a supersonic flow at zero angle of attack. A Pitot tube is inserted into the flow at a location on the back facing side of the airfoil. The pressure measured by the Pitot tube is 2.596 atm. On the same face before the Pitot tube the static pressure is measured 0.1 atm. Calculate the free-stream Mach number.

5. A uniform supersonic stream with $M_1 = 1.5$, $p_1 = 1700$ lb/ft^2, and $T_1 = 460^0$ R encounters an expansion corner which deflects the stream by an angle $\theta_2 = 20^0$. Calculate M_2, p_2, T_2, p_{02}, T_{02}, and the angles the forward and rearward Mach lines make with respect to the upstream flow direction.

6. Consider the adiabatic flow of air through a pipe of 0.2 ft diameter and 3 ft length. The inlet flow conditions are $M_1 = 2.5$, $p_1 = 0.5$ atm, and $T_1 = 520^0$ R. Assuming local friction coefficient equals a constant of 0.005, calculate the following flow conditions at the exit: M_2, p_2, T_2, and p_{02}.

7. At the inlet to the combustor of a supersonic combustion ramjet (SCRAMjet), the flow Mach number is supersonic. For a fuel-air ratio (by mass) of 0.03 and a combustor exit temperature of 4800^0 R, calculate the inlet Mach number above which the flow will be unchocked. Assume one dimensional frictionless flow with $\gamma = 1.4$, with the heat release per slug of fuel equal to $4.5 X 10^8$ ft.lb.

8. a) Show that for a shock to occur the upstream Mach number will have to be greater than equal to 1.

   b) Show that across an oblique shock the velocity component along the shock line remains constant.