Indian Institute of Engineering Science and Technology, Shibpur
BE (Aero) Part III 6th Semester Final Examination, 2014

High Speed Aerodynamics
(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 normal shock wave is standing in the test section of a supersonic wind tunnel. Upstream of the wave $M_1 = 3$, $p_1 = 0.5$ atm, and $T_1 = 200$ K. Find M_2, p_2, T_2 and u_2 downstream of the wave. Also show that these properties satisfy the Hugoniot equation for a calorically perfect gas.

2. a) Show that when heat is added pressure increases for a supersonic flow.

   b) Show that for subsonic inlet flow the effect of friction on the downstream flow is such that velocity increases.

3. An incident shock wave with wave angle = 30^0 impinges on a straight wall. If the upstream flow properties are $M_1 = 2.8$, $p_1 = 1$ atm, and $T_1 = 300$ K calculate the pressure, temperature, Mach number, and total pressure downstream of the reflected wave.

4. Consider a supersonic flow with an upstream Mach number of 4 and pressure of 1 atm. The flow is first expanded through an expansion corner with $\theta = 15^0$, and then compressed through a compression corner with equal angle $\theta = 15^0$ so that it returns to its original upstream direction. Calculate the Mach number and pressure downstream of the compression corner.

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

6. a) Consider the flow through a convergent-divergent duct with an exit-to-throat area ratio of 2. The reservoir presuure is 1 atm, and the exit pressure is 0.95 atm. Calculate the Mach numbers at the throat and at the exit.

   b) Consider the flow through a convergent-divergent duct with an exit-to-throat area ratio of 1.6. Calculate the exit-to-reservoir pressure ratio required to achieve sonic flow at the throat, but subsonic flow elsewhere.

7. a) Derive Euler's equation for conservation of linear momentum for an inviscid flow applicable to steady irrotational flow.

   b) Derive Crocco's theorem. Show that an inviscid steady flow behind a bow shock is rotational.

8. Show that following nonlinear equation is valid for transonic flow with small perturbations:

$$(1-M_\infty^2)\frac{\partial^2\phi}{\partial x^2}+\frac{\partial^2\phi}{\partial y^2}+\frac{\partial^2\phi}{\partial z^2}=M_\infty^2\,\frac{\gamma+1}{V_\infty}\,\frac{\partial\phi}{\partial x}\,\frac{\partial^2\phi}{\partial x^2}$$