Manipal Institute of Technology, Manipal

(A Constituent Institute of Manipal University)

V SEMESTER B.TECH (AERONAUTICAL ENGINEERING)

END SEMESTER EXAMINATIONS, NOV/DEC 2015

SUBJECT: AERODYNAMICS-II [AAE 305]

REVISED CREDIT SYSTEM

Time: 3 Hours MAX. MARKS: 50

Instructions to Candidates:

- Answer **ANY FIVE FULL** the questions.
- Aerodynamic flow property tables are allowed
- Missing data may be suitable assumed.

1A. Explain the features of numerical approach named Method of Characteristics. Write down this method's limitations and also draw the diagram of the Unit process procedure in this method. **(05)**

1B. Draw the diagram of left & right running intersection of opposite families with slip line (when left running deflection angle is bigger than right running curve) and pressure-deflection diagram at condition of deflection angle of left & right running curves are equal. **(03)**

1C. Why hypersonic flow boundary layer thickness is always greater than subsonic and supersonic flow? **(02)**

2A. Consider air enters a constant area duct at M_1=3.5, P_1=1.5atm, T_1=340K and ρ=1.225 kg/m^3. Inside the duct heat added per unit mass is q=4.2 x 10^5 J/Kg. Calculate the flow properties M_2, P_2, ρ_2, T_2, P_{02} and T_{02} at the exit of the duct. **(03)**

2B. Derive and prove that Mach number behind the normal shock is always subsonic and also derive the equations for downstream Mach number, density, pressure and temperature. **(05)**

2C. Why in supersonic flow any disturbances in the flow won't affect the upstream and downstream of the flow according to the linearized theory and also mention the equation for linearized pressure coefficient in supersonic flow. **(02)**

3A. Consider an infinitely thin flat plate at an angle of attack 12deg in a Mach 10 inviscid flow. Calculate the pressure coefficients on the top and bottom surface of the plate, the lift and drag coefficients and the lift-to-drag ratio by using **(03)**

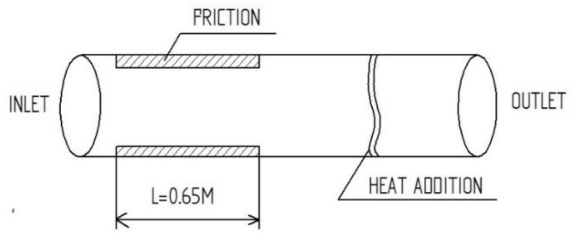
a) Exact shock and expansion waves theory
b) Newtonian theory
c) Compare both results

3B. Derive and prove Croccos theorem which states that flow field behind the curved shock is always rotational. **(03)**

3C. Describe the following **(04)**

a) Mach Reflection
b) Diagram of reflection of expansion wave on a pressure boundary
c) Comparison between convectional and supercritical airfoils
d) Newtonian flow

4A. Consider a constant area duct with diameter of 0.45m and the inlet conditions are M_1=2.5, P_1=1.2atm, T_1=276K. In the duct at the first section we are having constant friction coefficient (f=0.02) and the remaining duct we adding heat q=10^5 J/Kg. If then calculate the flow properties (M, T, P, ρ and P_0) at each sections (heat & friction). **(05)**



4B. Derive Area-Velocity relation and explain its properties. Draw the schematic diagram of supersonic wind tunnel and prove that diffuser throat section is always bigger than inlet throat section. **(05)**

5A. Consider a rocket engine and at its combustion chamber burning fuel with pressure and temperature of 36atm and 3800K respectively. The area of the throat is 0.38m^2. The area of the exit is designed such that the exit pressure exactly equals the ambient atmospheric pressure (2.4495 x 10^3 N/m^2) at standard altitude of 25km. Assume an isentropic flow through the rocket engine nozzle with an effective value of the ratio of specific heat γ=1.22 and R=520J/kg.K. if then **(05)**

a) Calculate the thrust of the rocket engine by using equation
$\mathrm{T} = \dot{\mathrm{m}} \cdot \mathrm{u_e} + (\mathrm{P_e} - \mathrm{P_\infty})\mathrm{A_e}$(where $\dot{m}$ = mass flow, $\mathrm{u_e}$, $\mathrm{P_e}$, $\mathrm{A_e}$ = velocity, pressure and area at the exit)
b) Calculate the area of the nozzle exit

5B. Derive and explain $\theta - \beta - M$ relation and define strong and weak solutions **(05)**

6A. Draw the graph of variation of linearized pressure coefficient with Mach number for both subsonic and supersonic flows. And also calculate the following numerical problem.(γ=1.4, R=287, ρ_∞=0.3648kg/m^3, T_∞=216.78K)
Consider a supersonic fighter aircraft in a Mach 2 flow at 11km altitude and planform area of the wing is 18.21m^2. The weight of the aircraft is 9400kgf. Assume that all the lift of the airplane comes from the lift of the wing. Calculate the angle of attach of the wing relative to the freestream (use linearized supersonic equation) **(05)**

6B. Derive the equation which explains linearized pressure coefficient depend only on the ‘x’-component of the perturbation velocity. **(03)**

6C. What is shock expansion method in hypersonic flow? And why it is not accurate in supersonic flows? **(02)**