Reg. No.										

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

V SEMESTER B.TECH. (AERONAUTICAL ENGINEERING)

END SEMESTER EXAMINATIONS, NOV/DEC 2019

SUBJECT: GAS DYNAMICS [AAE 3102]

REVISED CREDIT SYSTEM
(18/11/2019)

Time: 3 Hours MAX. MARKS: 50

Instructions to Candidates:

- Answer **ALL** the questions.
- Missing data may be suitable assumed.
- Compressible flow chart will be provided

1A. Express a relation for flow properties across shock wave only through thermodynamic properties and also represent a graphical way to determine the downstream properties of the flow. **(04)**

1B. **(04)**

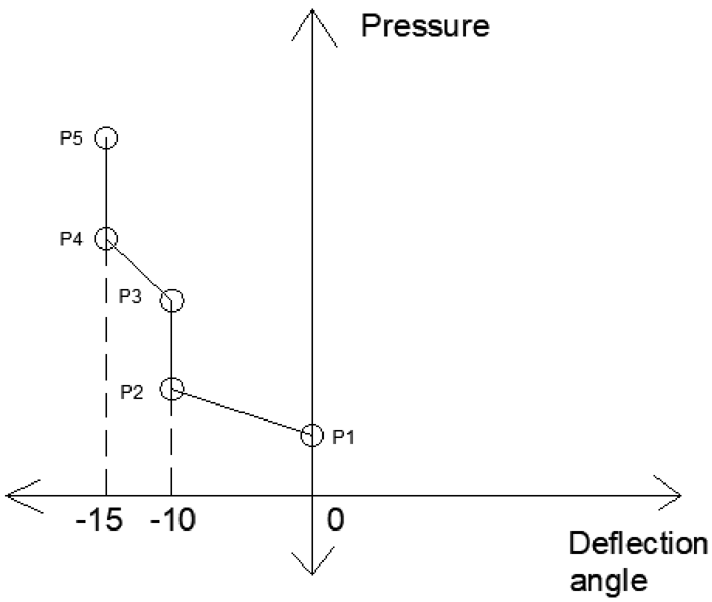

Figure - 1

Figure -1 representing a pressure-deflection diagram of a supersonic flow through various deflection surfaces with freestream M_1=3.2. Draw the complete schematic diagrams of corners and also calculate the Mach number at 5th region.

1C. Draw the schematic diagram of supersonic wind tunnel with its components **(02)**

2A. Derive an expression to show that the Mach number at any location in the duct is a function of the ratio of the local duct area to the sonic throat area. **(04)**

2B. Consider an airplane flying at standard sea level conditions and free stream velocity of 190km/hr. The flow accelerates over the wing and reaching a maximum velocity of 295km/hr some point on the wing. Calculate the following: **(04)**

a) What is the percentage pressure change between the maximum point and free stream? (assume incompressible flow)
b) What is the percentage density change between the maximum point and free stream? (assume compressible flow, R=287J/kg.K)
c) What will be the critical Mach number of this same wing if it is travelling at sea level?

2C. Distinguish the difference between Mach waves and Mach lines. Give example. **(02)**

3A. Define the exact solution of Method of Characteristics through equations and its procedures. Also mention a unit process and write down the streamline and Mach lines equations to determine any arbitrary point flow properties with known values of initial data line. **(04)**

3B. Consider a constant area duct with diameter of 0.24m and the inlet conditions are M_1=0.5, P_1=1.3atm, T_1=272K. In the duct at one section we are adding heat q=60700 J/kg, and after this, remaining duct having a constant friction with coefficient of f=0.004 (L=0.5m). Calculate the downstream flow properties (M, T, P, T_0, P_0) at each sections. **(04)**

3C. Write down the steps and procedures to determine downstream flow properties from the velocity potential equation with known value of velocity potential. **(02)**

4A. Derive an expression which relates incompressible flow over a given two-dimensional profile to subsonic compressible flow over the same profile. **(04)**

4B. Consider a flat plate with chord length of 1.2m. The free stream flow properties are M_1=2.8, P_1=1atm and T_1=270K. Using shock expansion theory, tabulate and plot following properties as functions of 'α' (angle of attack) by considering angle of attack varying as 2, 5 and 8 deg. **(04)**

a) Pressure on top & bottom surfaces
b) Temperature on top & bottom surfaces
c) L/D
d) Plot the results as comparison

4C. Write down linearized velocity potential equation and its limitations. **(02)**

5A. Define the followings **(04)**

a) Mach reflection
b) Thermal compressibility
c) Shock polar
d) Crocoss theorem

5B. Consider a supersonic fighter aircraft in a Mach 2.5 flow at 11km altitude and planform area of the wing is 18.21m^2. The weight of the aircraft is 10200kgf. Assume that all the lift of the airplane comes from the lift on the wing. Calculate the angle of attack of the wing relative to the freestream (use linearized supersonic equation) γ=1.4, R=287, P_∞=10kPa, T_∞=216.78K **(04)**

5C. Draw the diagrams of reflection of shocks from solid and free boundaries. **(02)**