Reg. No.										

MANIPAL INSTITUTE OF TECHNOLOGY
MANIPAL
A Constituent Institution of Manipal University

V SEMESTER B.TECH. (AERONAUTICAL ENGINEERING)

END SEMESTER EXAMINATIONS, NOV/DEC 2016

SUBJECT: GAS DYNAMICS [AAE 3102]

REVISED CREDIT SYSTEM
(26/11/2016)

Time: 3 Hours MAX. MARKS: 50

Instructions to Candidates:

- Answer **ALL** the questions.
- Missing data may be suitable assumed.

1A. In a supersonic wind tunnel how the diffuser efficiency is related to diffuser throat area (explain it through graphical representation)? Also write down how to eliminate the initial shock wave generated inside the wind tunnel. **(03)**

1B. Consider a rocket engine burning hydrogen and oxygen in the combustion chamber and here the temperature and pressure are 4000K and 30atm respectively. The molecular weight of the chemically reacting gas in the combustion chamber 18 and γ = 1.2. The pressure at the exit of the convergent – divergent rocket nozzle is 1.38 x 10^{-2} atm. The area of the throat is 0.35 m^2. Assuming a calorically perfect gas if then, Calculate : **(04)**

a) The exit Mach number
b) Exit velocity
c) The mass flow through the nozzle
d) Area at the exit.

1C. Explain the application of diffuser with an example for P_0=36.7atm, A/A^*=4.23, M_e=3. For a particular vale of $\frac{A}{A^*}$ there is only one possible shock free supersonic flow and there are multiple options for subsonic speeds. Explain the theory behind this. **(03)**

2A. Derive the jet propulsion thrust equation with the help of Momentum equation **(04)**

2B. Consider an airplane flying at standard sea level conditions and free stream velocity is 180km/hr. The flow accelerates over the wing and reaching a maximum velocity of 290km/hr. at some point on the wing. Then calculate the followings **(03)**

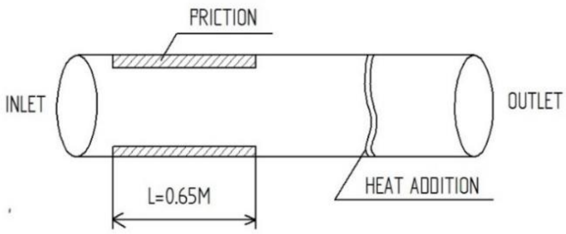
a) What is the percentage pressure change between this maximum point and free stream? (assume incompressible flow)
b) What is the percentage density change between this maximum point and free stream? (assume compressible flow, R=287J/kg.K)
c) What will be the critical Mach number of this same wing if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

2C. Derive the Hugoniot equation and explain how to determine the downstream properties of the flow from Hugoniot curve. **(03)**

3A. Describe the followings **(03)**

a) Croccos equation and theory
b) Mach divergence & supercritical airfoil
c) Calculation of Isothermal compressibility of air at 0.8atm

3B. Consider a constant area duct with diameter of 0.42m and the inlet conditions are M_1=2.3, P_1=1.25atm, T_1=275K. In the duct at the first section we are having constant friction coefficient (f=0.025) 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 section **(05)**



3C. What are the limitations of linearized perturbation velocity potential equation? And explain why these limitations? **(02)**

4A. Explain the features and procedures of numerical approach Method of Characteristics. Write down this method's limitations and also draw the diagram of the Unit process procedure in this method. **(05)**

4B. Consider a 16^0 half angle wedge with unit span at zero angle of attack in a Mach number 4 of air at standard conditions. Calculate: **(03)**

a) The pressure coefficient on the wedge
b) The drag coefficient (assume the pressure exerted over the base of the wedge, the base pressure is equal to the free stream pressure)

4C. Explain the importance of Ballistic and Lifting coefficients in reentry vehicle. Also write down if we consider both Earth's and Martian atmospheres then what kind of entry (lifting or ballistic) we prefer? Explain the reason. **(02)**

5A. Derive the equation $C_p = \frac{C_{po}}{\sqrt{1-M_\infty^2}}$, which showing the relation between incompressible and compressible flows with the help of transformed coordinates. **(04)**

5B. Consider an aircraft with planform area 25m^2 in steady level flight at Mach 2.5 at altitude of 11km. The weight of the aircraft is 11556kgf. 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 2.5m which almost equal to the mean chord length of the wing. If then, calculate the followings ($\rho_\infty = \frac{0.3648kg}{m^3}, T_\infty = 216.78K$) **(04)**

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

5C. In supersonic flights why do we use conical nose section? Also draw the graphical representation of Total pressure, Static pressure and Temperature with Mach number just behind the normal shock wave. **(02)**