Reg. No.										

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

V SEMESTER B.TECH. (AERONAUTICAL ENGINEERING)
END SEMESTER EXAMINATIONS, NOV 2019

SUB: AERODYNAMICS OF ROCKETS AND MISSILES [AAE 4001]

REVISED CREDIT SYSTEM
(20 / 11 / 2019)

Time: 3 Hours MAX. MARKS: 50

Instructions to Candidates:

- Answer **ALL** the questions.
- Missing data may be suitable assumed.
- Draw the diagrams only with the 'PENCIL'.
- DATA SHEETS will be provided from department.

1A. Write any four differences between Rocket and Missile. **(2)**

1B. With the help of neat diagrams explain different types of ramjet combustors. **(3)**

1C. A four stage rocket is used to put-up a satellite of 40 kg mass in a Low Earth Orbit (LEO). The approximate values of mass of propellant, mass of structure and jet velocity for each stage are given below: **(5)**

Stage	I-stage	II-stage	III-stage	IV-stage
Mass of propellant (Kg)	9000	3500	1700	260
Mass of structure including inert (kg)	1500	550	250	40
V_j (m/s)	2200	2400	2500	2750

Determine (i) The payload mass fraction of the total rocket (Satellite launch vehicle), (ii) Structural mass fraction of stage (iii) The ideal ΔV provided by each stage and the total ΔV. (iii) If the stage fires for a period of 50 seconds and the rate of mass depletion can be assumed to be constant, what would be the acceleration of the rocket at take-off?

2A. Explain the parts of Missile with neat diagrams. **(2)**

2B. Write the classification of Missiles based on Guidance systems. **(3)**

2C. An end-burning rocket uses a cylindrical double base propellant grain with a diameter of 200mm and generates a thrust of 350 N over a period of 300 sec. The thrust coefficient is 1.15. The characteristics of the propellant are; Density of propellant grain =1500 kg/m^3, Speed of Sound (a_{70})=4 mm/s, Choice of index (n)=0.5, Characteristic velocity (C*)=1500 m/s. Determine the length of propellant grain and throat diameter of the nozzle. **(5)**

3A. With the help of neat diagrams discuss the concept of wind effective flow direction rocket/missile body. **(2)**

3B. Derive the Ejection velocity **"V_j"** expression of gas from a high pressure chamber and also draw the conclusions **(3)**

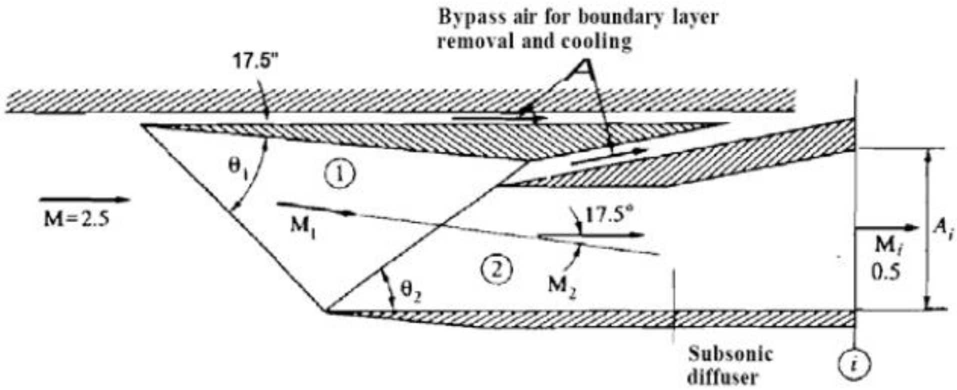
3C. A high pressure-fed liquid propellant rocket based on the gas generator cycle has a vacuum thrust of 735 KN and burn duration of 180 sec. The propellants used are Nitrogen Tetroxide (N_2O_4) and Unsymmetrical Dimethyl Hydrogen (UDMH). The specific impulse is **(5)**

2950 N-s/kg. The Mixture ratio is 1.87. The pressure in the thrust chamber of the rocket is 6MPa and the propellant supply pressure to the chamber is 7 MPa. N_2O_4 is stored in the propellant tank at a pressure of 0.4 MPa and UDMH is stored at 0.32 MPa. The densities of N_2O_4 and UDMH are 1400 Kg/m^3 and 790 Kg/m^3 respectively, at the temperature used in the rocket. Determine Power required driving N_2O_4 and UDMH pump.
If the efficiency of the pump is 66% and the turbine efficiency is 88% determine the mass flow rate through the turbine. The gas generator pressure and temperature can be assumed to be 3.5 MPa and 670 K respectively. The exit pressure of the turbine can be taken as 0.18 MPa. Assume specific heat of the gas at constant pressure as 1.9 KJ/kg K and the specific heat ratio of the gas Υ as 1.24.

4A. The altitude and orbit of a satellite are maintained using a number of small rockets housed in the satellite. The altitude and orbit corrections required during the lifetime of satellite are estimated to be 950 m/s. if the jet velocity of rocket is 2500 m/s and dry mass of satellite (Dry mass without the propellant being loaded in the satellite) is 800 kg, determine the mas of propellant required for the altitude and orbit corrections. **(2)**

4B. Explain the working principle with the neat diagrams of Gas generator cycle, Staged combustion cycle and Expander cycle. **(3)**

4C. A new supersonic missile is being designed for a flight Mach numbers 2.5 at an altitude where the ambient pressure and temperature are 9 KPa and 220 K respectively. The engine inlet configuration shown below allows for double oblique shock deceleration followed by a zone of subsonic deceleration. The Mach number is 0.5 at the engine inlet plane. **(5)**



Losses in the subsonic diffuser are neglected. Determine: (i) The Mach numbers M_1 and M_2 in the zones respectively shown on the drawing. (ii) The wave angles θ_1 and θ_2 also shown on the drawing. (iii) The overall stagnation pressure ratio $\frac{P_{0i}}{P_{0a}}$. (iv) The overall static pressure ratio $\frac{P_i}{P_a}$. (v) The velocity ratio $\frac{V_i}{V_2}$ for the subsonic diffuser. And (vi) The cross-sectional area A_i (m^2) at the engine inlet plane if the engine mass flow rate is 500 kg/s.

5A. Derive the performance and choice of feed system cycle in liquid propellant rocket. **(2)**

5B. Derive an expression for the analysis procedure for combustion instability. **(3)**

5C. Compare the specific fuel consumption of missiles turbojet and ramjet that are being considered for flight at Mach = 1.5 and 50,000 ft altitude. Ambient pressure and temperature 11.6 KPa and 205 K respectively. The turbojet pressure ratio is 12 and the maximum allowable temperature is 1400 K. For the ramjet the maximum temperature is 2500 K. For simplicity ignore aerodynamic losses in both engines. Conventional hydrocarbon fuels are to be used (Heating value of 45,000 KJ/Kg). Assume Υ = 1.4 and C_p=1.0 KJ / (kg. K) **(5)**