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: ROCKET PROPULSION [AAE 3103]

REVISED CREDIT SYSTEM

(01/12/2016)

Time: 3 Hours MAX. MARKS: 50

Instructions to Candidates:

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

1A. A liquid rocket engine with oxidizer to fuel ratio of 5:1 produces a thrust of 1 MN. The initial mass of rocket engine is 100,000 kg and its mass at burn out is 10,000 kg. the characteristic velocity C* and thrust coefficient C_F of the engine are 2386 m/s and 1.4 respectively. **(03)**

i. Find the mass flow rate of fuel.

ii. If initial velocity of engine is 2.5 km/s, find the velocity of rocket at burn out.

1B. Show that the maximum mass flow rate per unit area occurs at the throat of a C-D nozzle and find its value. **(04)**

1C. What are the desirable physical properties of liquid propellants? **(03)**

2A. A stationary two stage rocket with initial mass of 16,000 kg, carrying a payload of 1000 kg, is fired in a vertical trajectory from the surface of the earth. Both the stages of the rocket have same specific impulse, Is, of 300 s and same structural coefficient of 0.14. The acceleration due to gravity is 9.81 m/s^2. Considering both stages with same payload ratio, find the terminal velocity attained by the payload. **(03)**

2B. Derive rocket equation. **(03)**

2C. What is burn rate? How is it influenced by the combustion chamber pressure and temperature? **(04)**

3A. The following measurements were made in sea level of a solid rocket motor: **(06)**
Burn rate = 40 sec, Initial mass before test = 1210 kg,

Mass of motor after test = 215 kg, Average thrust = 62,250 N, Chamber pressure = 7 MPa, Nozzle exit pressure = 0.07 MPa, Nozzle throat diameter = 0.0855 m, Nozzle exit diameter = 0.2703 m. Determine:

i. Propellant flow rate, nozzle exit velocity, Characteristic velocity, effective exhaust velocity and specific impulse at sea level.
ii. Find Specific impulse at 1000 and 25,000 m altitude.

[Take P_{1000m} = 0.0898 MPa and $P_{25,000m}$ = 0.00255 MPa]

3B. A solid rocket motor is designed with a cylindrical end-burning propellant grain of length 1 m and diameter 32 cm. The density of the propellant grain is 1750 kg/m^3. The specific impulse of the motor is 190 s and the acceleration due to gravity is 9.8 m/s^2. Find the thrust produced by the rocket motor if the propellant burns for a period of 150 s. **(02)**

3C. Describe the following terms related to solid rocket motor: **(02)**

i. Double base propellant
ii. Additives
iii. Ablatives
iv. Binder

4A. With a neat sketch, explain the working of nuclear thermal rockets. **(03)**

4B. What are the parts of an Ion propulsion system? Explain. **(03)**

4C. What are the different thrust vector control methods in rocket engines? **(04)**

5A. Write a brief note on the following: **(03)**

i. Inlet buzz
ii. Mixed compression inlets

5B. Explain briefly about combustion in Ramjet and Scramjet engines. **(03)**

5C. Hot combusted gas enters the SCRamjet nozzle at a static temperature (T_1) of 2500 K and static pressure (P_1) of 300 kPa and Mach number (M_1) of 2. The flight speed is (V_∞) 2000 m/s at an altitude where the ambient pressure (p_∞) is 1000 Pa, and the area of the nozzle entrance is (A_1) 0.01m^2. **(04)**

If the nozzle exit area is (A_e) 0.12 m^2 and flow throughout the nozzle is isentropic with R = 289 J/kgK and γ=1.25, find:

i. Total temperature and pressure in the nozzle.
ii. Mass flow rate through the nozzle.
iii. Exit Pressure and Mach number
iv. Thrust produced by the nozzle