Roll No: --------------------------

UNIVERSITY OF PETROLEUM AND ENERGY STUDIES

End Semester Examination, December 2017

Program: B. tech ASE	**Semester – VII**
Subject (Course): Rocket Propulsion	**Max. Marks : 100**
Course Code : ASEG 324	**Duration : 3 Hrs**
No. of page/s: 2	

Instructions- Read all the below mentioned instructions carefully and follow them strictly

1) ATTEMPT ALL THE PARTS OF A QUESTION AT ONE PLACE ONLY.
2) Internal choice is given for question number 10.
3) Assume suitable data if required

Section A [5 X 4 = 20 marks]

1) Explain the need of solid propellants in booster rocket with illustrative example.

2) Derive the following expression for rocket

a. Propulsive efficiency = $\left[\frac{2\,\sigma}{1+\sigma^2}\right]$

b. Thermal efficiency = $\left[\frac{C^2}{2Q_R}(1+\sigma^2)\right]$

3) Compare the solid propellant, liquid propellant and electrical rocket in terms of basic performance parameters.

4) How are regressive, neutral and progressive burning of the solid propellant grain achieved? explain with the aid of diagrams.

Section B [10 X 4 = 40 Marks]

5) Explain the solar and electrical rocket engine with suitable sketches.

6) Established the relation between specific impulse, propellant mass fraction and burn out time for power and coasting flight.

7) A) Derive the rocket equation for velocity increment of multistage rocket. [5 marks]

B) Derive the nozzle efficiency from steady flow energy equation and also define the effect of nozzle friction. [5 matrks]

8) A missile has maximum flight speed to jet speed ratio 0.56 and specific impulse 250 seconds. Determine for a burn out time 12 seconds.

a) Effective jet velocity b) Mass ratio and propellant mass fraction

c) Maximum flight speed and d)Altitude gain during powered and coasting flights.

Section C [20 X 2= 40 marks]

9) Following date are given for a Rocket engine:

Combustion chamber temperature	= 3500 K
Combustion chamber pressure	=38 bar
Oxidizer flow rate	= 41.67 kg/s
Mixture ratio	= 5
Specific heat ratio	= 1.3
Gas constant (R)	= 287 J/kg K
Nozzle expansion ratio	= 75
Divergence cone angle	= 32 degree
Convergence half cone angle	= 55 degree
Hoop stress of the material	= 60 MPa

If the expansion in the rocket nozzle takes place to the ambient pressure 583.59 N/m^2 calculate

a) Nozzle throat area b) thrust c) thrust coefficient d) characteristics velocity e) exit velocity of exhaust gas f) divergent length of nozzle g) convergent length of nozzle h) thrust chamber length i) thickness of thrust chamber j) are at entry and exit of nozzle

10) A new research cryogenic engine is to be designed to operate with liquid methane and LOx. Methane will play the role of hydrogen in conventional engine. Assume that the chamber pressure is 20 MPa and the reactants reach a temperature of 298 K while entering the thrust chamber. The combustion takes place at stoichiometric condition. The nozzle used for SSME is planned for this engine as well. It has a throat diameter of 0.26 m and nozzle expansion ratio of 77.5. Calculate: Chamber temperature and the average molecular weight of the product gases, Characteristic velocity assuming specific heat ratio = 1.22, Vacuum thrust, vacuum specific impulse Thrust Coefficient Optimum flying altitude.

The mass flow rates of methane and oxygen into the engine at steady operation

Data:

Heats of formation at 298K (kJ/mol):

CO_2= -393.978, H_2O= -241.997, CH_4= -74.943

Specific heats (kJ/kmol K):

CO_2= 58.29, H_2O= 47.1

(Assume any other data if required)

Or

11) Write down the selection criteria for space vehicle in details and also explain the idealized selection process with sketches.

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Section A [4 X 5 = 20 marks]

1) Explain the influence of following parameters on the performance of solid propellant rocket: chamber pressure, propellant exposed surface area, initial grain temp and burning rate
2) Write down the properties of liquid propellant.
3) Explain the gas feed liquid propellant rocket engine and hybrid rocket engine.
4) Plot the burning phenomenon for the following shape in the solid propellant rocket.
   End burning grain structure, internal burning tube, internal burning tube and slot, external burning rod. And internal burning multiple perforation
5) Calculate the burning rate and burning surface area for a propellant from given data.
   Propellant area ratio = 2000
   K_2 (fraction depends on grain and initial temp) = 50
   K_1 (fraction depends on chemical composition) = 8
   N_2(Constant) = 1.2
   N_1 (Constant) = 0.6

Section B [10 X 4 = 40 Marks]

6) Explain the following in details

   a) Monopropellants b) hypergolic propellants c) bipropellants d) UDMA e) RFNA

7) Explain briefly the following terms in solid propellant rocket

   a) Linear burning rate b) combustion rate c) propellant area ratio d) equilibrium combustion pressure e) cigarette burning

8) An advanced fighter engine operating at Mach 0.8 and 10 km altitude (ambient temperature of 223.15 K) has a following data, and uses a fuel with calorific value 42800 kJ/kg Thrust =50 KN air mass flow = 45kg/s, and fuel mass flow =2.65kg/s. Find the specific thrust, TSFC, exit velocity, thermal propulsive and overall efficiencies

9) A) A rocket nozzle has a throat area of 18 cm^2 and combustion pressure of 25 bar. If the specific impulse is 127.42 seconds and weight flow rate 44.1454 N/s. Determine

a) Thrust coefficient	b) Propellant weight flow coefficient	
c) Specific propellant consumption	d) Characteristics velocity	**[5]**

B) Calculate adiabatic flame temperature of mixture of liquid methane and liquid oxygen from following given data. where reference temperature is 298 K. **[5]**

Heats of formation at 298K (kJ/mol):

CO_2 = -393.978, H_2O= -241.997, CH_4= -74.943

Specific heats (kJ/kmolK):

CO_2= 58.29, H_2O= 47.1

Section C (20 X 2 = marks)

10) A) Starting from isentropic relation for an ideal exhaust nozzle with discharge coefficient of unity prove that its thrust coefficient is given by **[15]**

$$C_F = \gamma\sqrt{\frac{2}{\gamma-1}}\left(\frac{2}{\gamma+1}\right)^{(\gamma+1)/2(\gamma-1)}\sqrt{1-\left(\frac{Pe}{Po}\right)^{(\gamma-1)/\gamma}}$$

B) Determine the values of thrust coefficient for a gas with specific heat ratio is 1.2 at pressure ratio 2 and 5. **[5]**

Or

Following data are given for a four stage rocket:

Stages	Mass of propellant (Kg)	Mass of structure (Kg)	Payload mass (Kg)	Jet velocity (m/s)
I (Booster)	10000	1700	50	2250
II	4500	800		2450
III	1900	350		2550
IV	360	50		2850

Determine:

a) Velocity increment for each stages b) total velocity require for initial thrust

c) Propellant mass fraction d) Payload mass fraction e) structural mass fraction at each stages f) initial acceleration required if time of burning of booster rocket is 50 sec.

11) A) Define the selection criteria for Rocket engine in details with suitable sketch. **[10]**

B) Air at 1 Mpa and 600^0 C enters a convergent nozzle with a velocity of 150 m/s. determine mass flow rate through the nozzle when the nozzle throat area is 50 cm^2 and the back pressure is 1) 0.7 Mpa 2) 0.4 Mpa **[10]**