Assignment 4

1 A cryogenic rocket, called as Vinci, working on the expander cycle is proposed for the upper stage of the Ariane launch vehicle. In this rocket, hot hydrogen gas, generated in the regenerative cooling passage of the thrust chamber, drives two turbines installed in series. The first turbine drives the liquid hydrogen pump, while the second turbine drives the liquid oxygen pump. The two separate turbo-pumps, each equipped with an inducer, pressurize the propellants. Assuming the power of the hydrogen and oxygen turbo-pumps as 2500 kW and 400 kW, and their speeds as 90,000 RPM and 19,000 RPM respectively, determine the specific speed of the pumps. You can assume the efficiencies of the both pumps as 0.6. The thrust developed by the rocket in vacuum is 180 kN and the vacuum specific impulse is 4560 $\frac{N\,s}{kg}$. The mixture ratio in the thrust chamber can be taken as 5.8. You can assume the density of liquid hydrogen as 75 kg/m^3 and the density of liquid oxygen as 1150 kg/m^3.

2 Determine the heat to be transferred in the regenerative coolant passages of a cryogenic rocket using liquid hydrogen and liquid oxygen and operating in the expander cycle, given the following data. A single turbine is configured to drive both the LH_2 and LOX pumps :

Thrust: 10 kN
Mixture ratio: 5
Specific impulse: 4500 $\frac{Ns}{kg}$
Temperature of liquid hydrogen in storage tank: 20 K
Supply pressure of LH_2 and LOX to the pumps from the tanks: 0.25 MPa
Outlet pressure of LH_2 pump: 4.6 MPa
Outlet pressure of LOX pump: 3.6 MPa
Pressure drop in the regenerative coolant passage : 0.7 MPa
Temperature at hydrogen injection into thrust chamber = temperature of hydrogen at outlet from turbine : 150 K
Efficiency of the turbo-pump system: 0.65
Specific heat of liquid hydrogen: 9.7 $\frac{kJ}{kgK}$
Specific heat of gaseous hydrogen: 14.9 $\frac{kJ}{kgK}$
Density of liquid hydrogen: 75 kg/m^3
Density of liquid oxygen: 1150 kg/m^3
Latent heat of vaporization of liquid hydrogen: 446 kJ/kg
Boiling temperature of liquid hydrogen in coolant passage : 23 K

State any assumptions made.

3 A cryogenic liquid propellant rocket using liquid hydrogen for fuel and liquid oxygen for oxidizer works on a gas generator cycle. The overall mixture ratio of the rocket is 4.0 while the mixture ratio in the gas generator is 0.6. The temperature of the hot gases in the gas generator (corresponding to the mixture ratio of 0.6) is 700 K. The pressure ratio across the turbine is 3. Find the fraction of the propellants required to be supplied to the gas generator and the mixture ratio in the main thrust chamber?

You can assume the efficiency of the turbo-pump system as 0.6. The specific heat ratio and molecular mass of the products of combustion of the gas generator are 1.35 and 3.3 kg/kmole respectively. The densities of liquid hydrogen and liquid oxygen are 75 and 1140 kg/m^3 respectively. The pressure rise across the liquid hydrogen pump and liquid oxygen pump can be taken to be 5 MPa.

4 In the above problem, if the specific impulse in the main thrust chamber is 4450 Ns/kg and the specific impulse of the gases expanded from the turbine exhaust is 2500 Ns/kg, what is the specific impulse of the gas generator fed cryogenic rocket.

5 Determine the volume of gas bottle required to pressurize the propellant tanks containing UDMH and N_2O_4 in a high thrust liquid propellant rocket operating in a gas generator cycle. The gas used is helium and is stored in the gas bottle at a pressure of 32 MPa and a temperature of 300 K. The volume of the UDMH tank and N_2O_4 tank is the same and is 15,400 liters. Both fuel and oxidizer are supplied to the pumps at a rate of 110 liters/s. The supply pressure of UDMH is 0.37 MPa while that of N_2O_4 is 0.34 MPa. Two separate pressure regulators are used which reduce the pressure of the helium gas from the gas bottle pressure to 0.37 MPa for UDMH tank and 0.34 MPa for the N_2O_4 tank.

You can assume the UDMH and N_2O_4 tank to be completely filled with propellants. The pressure loss in the feed-lines and flow control components can be neglected.

6 a.) Calculate the combustion volume of a thruster burning Liquid Oxygen (LO_2) and Kerosene at a pressure of 5 MPa and generating a thrust of 60 kN. The actual thrust coefficient can be taken as 1.4 and the characteristic velocity C* as 2800 m/s. The characteristic length (L*) for a kerosene - LO_2 thruster can be assumed as 1.7 m.

b.) If the mixture ratio in the thruster is 2, determine the volumetric flow rates of LO_2 and kerosene. You can take the specific gravity of LO_2 and kerosene as 1.14 and 0.8 respectively.

c.) If like doublet injection elements with sharp-edged orifices of diameters of 1 and 1.14 mm are used for LO_2 and kerosene, find the number of injection elements required? You can assume the discharge coefficient of the orifices as 0.64 and the supply pressure to the injector as 6 MPa.

7 a. A liquid propellant rocket uses hydrazine for fuel and N_2O_4 for oxidizer. The rate of hydrazine injection is 4 g/s and that of N_2O_4 is 6 g/s.
If the mean diameter of hydrazine and N_2O_4 droplets formed in the spray is 0.3 mm and the mean axial velocity of droplets is 50 m/s, determine:

i. Mixture ratio at injection

ii. Mixture ratio of the vaporized propellant.

You can take the length of the combustion chamber as 50 cm. The evaporation of the droplets can be assumed to be given by the law: $d^2 = d_0^2 - \lambda t$, where the evaporation constant λ is 3 mm^2/s for hydrazine droplets and 5 mm^2/s for N_2O_4 droplets.

b. If the characteristic velocity C* in m/s is expressed in terms of mixture ratio by the expression:

$C^* = 2000 - 100(|R - 1.4|)$,

find the C* efficiency due to incomplete vaporization.

c. In the above problem, if the distribution of mixture ratio in the combustion chamber provides four streams with a core having 40% of the propellant flow and surrounded with annular streams having 30%, 20% and 10% of the total propellant flow, determine the mixture ratio of the stream adjacent to the chamber walls? The mixture ratio of the core is 1.6. The mixture ratio of the streams conveying 30% and 20% of the flow is 1.5 and 1.4 respectively.

Determine the C* efficiency due to mixture ratio distribution.

d. What is the net C* efficiency?

8 A liquid propellant rocket developing a thrust of 500 N uses MMH and N_2O_4 for the propellant at a mixture ratio of 1.65. The chamber pressure is 0.7 MPa. The value of the characteristic velocity C* of the propellant at the above chamber pressure and mixture ratio of 1.65 is 1800 m/s. The thrust coefficient C_F of the rocket is 1.5.

Determine the following:

i. Throat area of the nozzle

ii. Mass flow rate of MMH and N_2O_4 .

iii. The diameter of the injection holes to be provided in the injector for the MMH and N_2O_4 if 10 doublet injector elements are used. The injection pressure of MMH and N_2O_4 is 1 MPa. The discharge coefficient of the orifices is 0.95.

The density of MMH is 868 $\frac{kg}{m^3}$ and the density of N_2O_4 is 1400 $\frac{kg}{m^3}$.

9 Determine the power in kW of the turbine of a cryogenic liquid propellant rocket using liquid hydrogen and liquid oxygen operating on a staged combustion cycle given the following data:

Chamber pressure: 20 MPa

Injection pressure (for both propellants): 1.4 times the chamber pressure

Pressure at which propellants are stored in the propellant tank: 0.4 MPa

Mass flow rate of liquid hydrogen: 40 kg/s

Mass flow rate of liquid oxygen: 160 kg/s

Turbo-pump efficiency: 60%

Density of liquid hydrogen: 75 $\frac{kg}{m^3}$; Density of liquid oxygen: 1150 $\frac{kg}{m^3}$.

You can neglect the pressure drop in the propellant supply lines