Liquid Bipropellant Rocket Cycles and Components

• Homework solutions should be neat and logically presented, see format requirements at seitzman.gatech.edu/classes/ae6450/homeworkformat.html. Show ALL work in this format. If you use equations from the notes, class textbook or another book, please cite the reference.

• Remember the requirement to draw some IMPLICATIONS/CONCLUSIONS from the results of each problem.

• For these problems you may wish to use a numerical tool (spreadsheet, Matlab, etc.) to perform the analysis. If so, you still need to include in your write-up your analysis approach, the equations solved/used, etc. (this requirement can not be met by inserting your code in the homework).

You have been asked to assist with the preliminary design of a high-performance staged-combustion cycle for a rocket engine with the following attributes:

- methane fuel and LOX oxidizer;
- required sea-level thrust = $5.25 \times 10^5$ lbf (~2.34 MN);
- overall O/F (mass) ratio = 3.3;
- operating pressure for main combustion chamber = 160. atm;
- adiabatic flame temperature in main combustion chamber = 3660 K
- nozzle with expansion ratio of 30.0 and divergence correction factor of 0.985

1. Rocket Performance

Based on the above design inputs, estimate the following rocket parameters. Do not assume the nozzle exhaust is calorically perfect, rather use a shifting equilibrium assumption to calculate the nozzle performance (this will require using appropriate chemical equilibrium software).

a) required mass flowrates for the fuel and for the oxidizer;

b) required nozzle throat diameter (assuming a circular cross-section);

c) specific impulse of the engine at both sea-level and in vacuum.

For the remaining parts of this exercise, use your results from part 1), and you should assume the following:

- Preburner and Cooling
  - the staged combustion cycle uses a single oxygen-rich preburner, with all the oxidizer going through the preburner (this is sometimes known as an oxygen-rich staged combustion ORSC cycle);
  - the preburner injector pressure drops are 6% (oxidizer) and 12% (fuel) of the preburner operating pressure;
• Preburner exit conditions can be determined as a function of the preburner O/F ratio using polynomials of the form
  \[ y = a + b(O/F) + c(O/F)^2 + d(O/F)^3 \]
  with the coefficients for various parameters given in the table below (note: the results are only valid for the range O/F = 29-60).

<table>
<thead>
<tr>
<th></th>
<th>a</th>
<th>b</th>
<th>c</th>
<th>d</th>
</tr>
</thead>
<tbody>
<tr>
<td>T (K)</td>
<td>3656.45</td>
<td>-127.145</td>
<td>2.0653</td>
<td>-0.012926</td>
</tr>
<tr>
<td>MW</td>
<td>29.9381</td>
<td>0.045404</td>
<td>-3.2397×10^{-4}</td>
<td>----</td>
</tr>
<tr>
<td>γ</td>
<td>1.18587</td>
<td>0.0032925</td>
<td>-1.6041×10^{-5}</td>
<td>----</td>
</tr>
</tbody>
</table>

• The portion of the fuel that is not used in the preburner is used to regeneratively cool the main combustion chamber;
• The pressure drop in the regenerative cooling loop is 18% of the main chamber pressure;

• **Propellant Storage and Pumps**
  • Both the fuel and oxidizer storage tanks are maintained at 3 bar;
  • The fuel enters its pump at 106 K, the oxygen enters its pump at 85 K;
  • The pump efficiencies are 80% (oxygen pump) and 75% (fuel pump);
  • Downstream of each pump is a control valve with a 5 bar pressure drop;

• **Turbine**
  • The cycle employs a single-stage turbine to run both fuel and oxidizer pumps on a single shaft;
  • The turbine efficiency is 70%;
  • In addition to the pumps, the turbine must also supply a 650 kW draw to run auxiliary equipment;
  • The shaft efficiency (connecting turbine to pump+auxiliary units) is 98%;

• **Main Combustor**
  • The main combustor injector pressure drops are 18% of the main chamber pressure;

• **Line Pressure Losses**
  • Negligible for this exercise;

• **Throttling Orifice**
  • Excess pressure in the fuel flow (above what is needed to overcome the main chamber injector and regenerative cooling pressure drops) is removed with a throttling orifice placed before the fuel enters the regenerative cooling system.
  • The orifice has a discharge coefficient of C_d=0.6
2. **Pressure and Power Requirements**
Determine the following:

a) the required fuel and oxidizer pump discharge pressures for two turbine inlet temperatures: 950 K (historic limit range) and 1350 K (very high temperature alloy);

b) the required turbine output power per unit mass flow rate (in kJ/kg) for the two turbine inlet temperatures;

c) the diameter of the required throttling orifice (assuming a circular orifice) for the 1350K turbine inlet temperature.

3. **Pump Design**
Determine the following for the design using the 1350K turbine inlet temperature:

a) assuming the maximum allowable suction specific speed for a pump using a high performance inducer is 60,000 (in US units), what is the maximum allowable RPM for our ungeared turbopump;

b) based on this maximum RPM, what is the specific speed of the oxygen and fuel pumps - AND determine if we can use a centrifugal-type pump for either or both of them;

c) estimate the outer impeller diameter and tip blade speed required for the fuel pump to see if the blade speed exceeds the standard range for steel alloys, e.g., 300-400 m/s; you may find helpful the specific diameter versus specific speed figure presented in class and available on the course web site;

d) repeat c) for the oxidizer pump.

4. **Turbine and Main Combustor Design**
Determine the following for the design using the 1350K turbine inlet temperature and the maximum RPM determined in Part 2:

a) if we intended to use a single-stage impulse turbine for this rocket, what would be the mean radius blade speed and the mean radius of the rotor;

b) assuming the turbine rotor tip speed can not exceed 300 m/s (since our turbine inlet temperature is rather high) and the nozzle exit angle can not exceed 70\(^\circ\), determine if a single-stage impulse turbine can be used for this engine design;

c) estimate the volume required for the (main) combustion chamber.