AE 330: Aerospace Propulsion – Tutorial 3

Rocket Performance Parameters

1. Consider a liquid propellant rocket engine having a characteristic velocity of $1850 m/s$, chamber pressure of $5 MPa$, throat diameter of $20 cm$, nozzle expansion ratio of 20. Let the product gases have $\gamma = 1.2$. Calculate the following at vacuum and sea level: mass flow rate, thrust, specific impulse. **Note:** Verify whether the flow separates in the nozzle, and take it into account in your calculations.

2. Consider the two separation criteria mentioned in the class. Attached is a plot of the variation of thrust coefficient with area ratio and ambient pressure. Update the plot with the following:

   (a) At area-ratios, 1, 10, 100, 1000, compute the thrust coefficients for optimum thrust, and mark them on the plot.

   (b) For the given ambient pressures, compute the thrust coefficients for optimum thrust, and mark them on the plot.

   (c) Do the above points lead to a smooth curve representing thrust coefficient for optimum thrust?

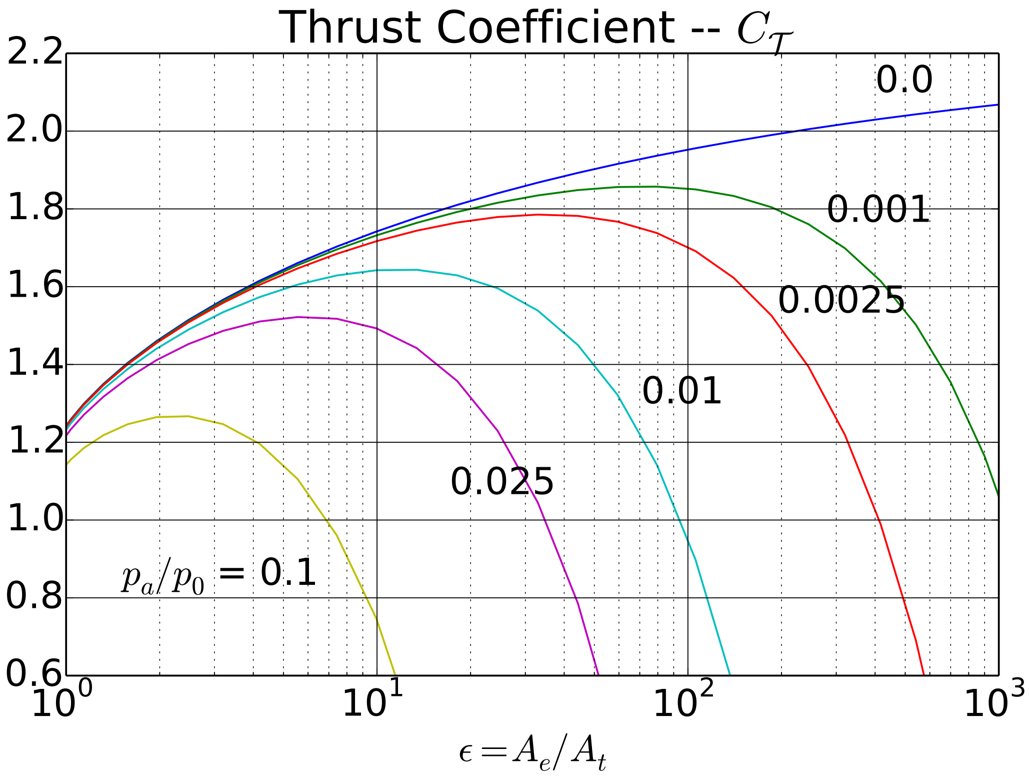
   (d) Consider nozzle flow separation near the exit. For a given ambient pressure ratio, find the area ratio beyond which flow will separate in the nozzle. Calculate the thrust coefficient for these conditions.

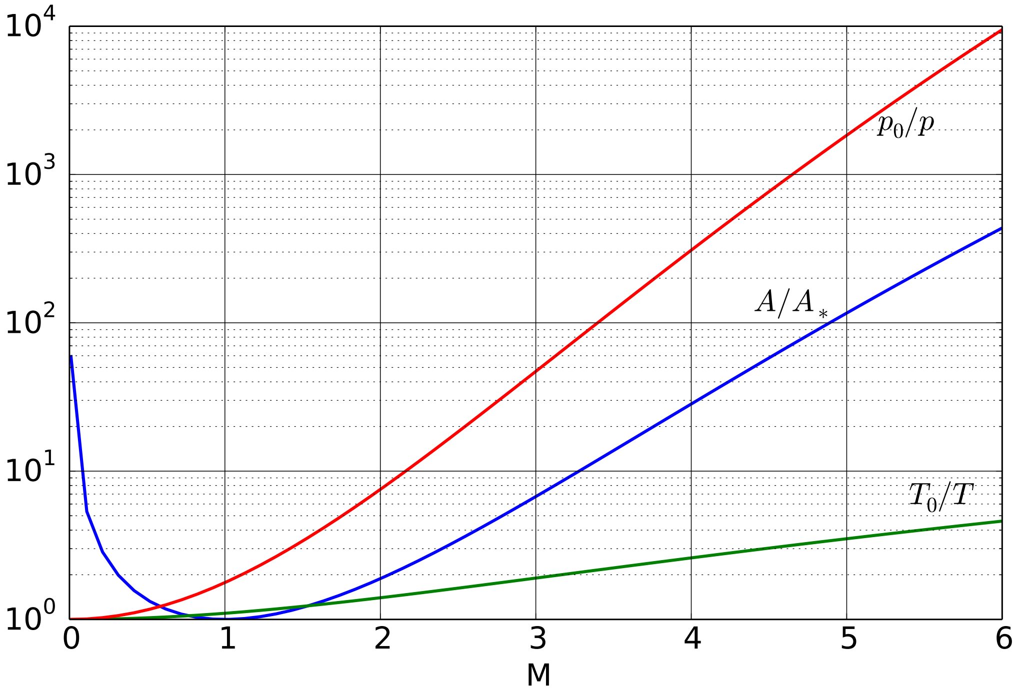
   (e) Using the earlier method for optimum thrust, arrive at the curves for the onset of separation. Compare the two correlations discussed in the class by drawing their respective separation cut-off curves.

3. A rocket nozzle is expected to conform to the following conditions: chamber pressure of $20 atm$, chamber temperature of $2860 K$, mean molecular weight of product gases of $22 gm/mol$, ideal specific impulse of $230 s$ at design altitude, and a thrust of $1300 N$ at sea level. Assume that the product gases have a specific heat ratio of 1.2. Calculate the nozzle throat and exit areas, respective diameters, actual exhaust velocity, and actual specific impulse.

4. An ideal rocket is flying at an altitude where the ambient pressure is one-hundredth of its chamber pressure. At this altitude, it has a characteristic velocity of $1220 m/s$, a mass flow rate of $73 kg/s$, a thrust coefficient of 1.5, and a nozzle throat area of $0.0248 m^2$, compute the effective exhaust velocity, chamber pressure. Also, calculate the thrust, specific impulse at vacuum, sea-level, and the design altitude. Estimate the design altitude, and the altitude below which there is flow separation.

5. The following information is given for a rocket: product gases have molecular weight of $24 kg/k-mol$, $\gamma = 1.2$, chamber pressure and temperature of $2 MPa$ and $2900 K$ respectively, throat area of $0.00050 m^2$, and an ambient pressure of $0.2 bar$ at design altitude. Determine the following: flow velocity & density at throat, mass flow rate, specific impulse/thrust at sea level, design altitude and vacuum. Account for the possibility that the flow may be separated at low altitudes.





$$p_a = p_{sl}\,\mathrm{exp}\left(-h/h_L\right) \qquad h_L = 8435m$$

$$c^\star = \sqrt{\frac{RT_0}{\gamma}\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{\gamma-1}}}$$