Note (for any problem that needs it) Universal gas constant, $\hat{R}$ = 8314 *J/kmol-K*.

(Picture from http://www.boeing.com/defense-space/space/propul/SSME.html)

1. A rocket nozzle is designed to produce 3.82 *MN* of thrust in vacuum, using an H_2-O_2 propellant combination that has a vacuum specific impulse, $I_{sp} \equiv \Im / \dot{m} g_o$ = 480 *s*. The mean molar mass of the propellant is 15 *kg/kmol*, and the ratio of specific heats γ = 1.18. The stagnation pressure in the combustion chamber is 19.8 *MPa*, and the stagnation temperature is 3400 *K*. Assume an ideal nozzle.

a. Estimate the mass flow rate of propellant and the throat area.

b. If the nozzle has an exit area ratio A_e/A_t = 72, estimate the pressure at the exit plane, and the sea level thrust produced by the engine.

2. A rocket engine uses a propellant with a mean molar mass of 21.9 *kg/kmol* and ratio of specific heats γ = 1.23. The stagnation pressure at the inlet to the nozzle is 2.15 *MPa* and the stagnation temperature is 2860 *K*. The rocket nozzle is to be designed for operation at sea level (p_a = 101 *kPa*). The design thrust is 1500 *N* with a nozzle efficiency η_n = 0.96.

(a) Calculate the propellant mass flow rate needed to obtain the design thrust.

(b) Calculate the required throat area.

(c) Calculate the required nozzle exit area.

3. A rocket engine uses a H_2-O_2 mixture as a propellant. The mean molar mass of the propellant is 11.58 *kg/kmol* with ratio of specific heats γ = 1.20. The *stagnation* pressure at the throat of the nozzle is 8.26 *MPa* and the *static* temperature at the throat is 3305 *K*. The throat area A_t = 750.4 cm^2. The ratio of the exit plane area to the throat area is 39.8 and the static pressure at the exit plane of the nozzle is 18.1 *kPa*.

(a) Calculate the mass flow rate of propellant through the nozzle.

(b) Calculate the vacuum thrust of the rocket motor.

(c) If the flow is assumed isentropic from combustion chamber to the nozzle throat, and the ratio A_c/A_t = 3.24, where A_c is the area of the combustion chamber, estimate the Mach number at the exit of the combustion chamber.