MAE 150R
Rocket Propulsion Systems
Assignment 4: Due Monday, May 15, 2000
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1. Consider a liquid propellant tank that is pressurized at p_{tank}. The liquid in the tank is driven through a centrifugal pump, across which there is a pressure rise Δp_{pump}, and then through a liquid injector with pressure drop Δp_{inj} corresponding to a liquid injection velocity u and liquid density ρ before the liquid enters the rocket's combustion chamber (with chamber pressure p_c). Since we are dealing with flowing liquids in this system, assume that the stagnation and static pressures of the liquids are essentially the same.

   a.) If the discharge coefficient of the injector is $C_d = 1$, write down the expression for the pressure drop Δp_{inj} in terms of ρ and u.

   b.) Write down the expression for the chamber pressure p_c in terms of p_{tank}, Δp_{pump}, ρ, and u.

2. (Problem 13.1 in Hill and Peterson) Consider the liquid pumps necessary to supply a liquid propellant rocket engine with a thrust of 6.7 MN. The engine consumes liquid oxygen and liquid fuel (RP1) in the ratio 2.5:1 by mass flux. The chamber pressure is 10 MPa, the flame temperature is 3300K, and the engine's specific impulse is 242 sec. The densities of liquid oxygen and RP1 are 1140 and 750 kg/m^3, respectively. The injection velocity of each of the liquids is 50 m/s, and the discharge coefficient for each liquid injector is $C_d = 1$. The propellant tank pressure is 0.35 MPa and the efficiency of each of the liquid oxygen and liquid RP1 tanks is $\eta_p = 0.65$ (see p. 624 for the definition of η_p).

   Show that the TOTAL shaft power (to drive BOTH liquid pumps) is 47,640 kW in this problem.

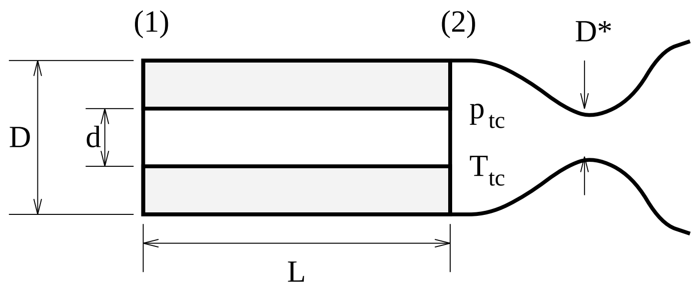
3. (Problem 12.11 in Hill and Peterson) Propellant tests are performed which, for a particular solid propellant, indicate a regression rate r_b of 0.06 cm/s when the chamber pressure p_c is 5 atm and a regression rate of 0.36 cm/s when the chamber pressure is 100 atm.

   a.) Compute the coefficient a and the exponent n for this propellant according to St. Robert's Law for solid propellants.

   b.) If this propellant is used in a rocket engine with a chamber pressure of 7 MPa and chamber temperature 2500K, determine the ratio of the burning area A_b to the throat area A_* that is required for steady combustion. The propellant density is $\rho_p = 1900 kg/m^3$ and the ratio of specific heats for the gaseous combustion products is $\gamma = 1.25$, with a molecular weight of the gases of 20.5.

   c.) If the burning area ratio A_b/A_* is increased from the value in a.) by 0.7 % per second, how LONG would it take for the chamber pressure to double?

4. (Problem 12.14 in Hill and Peterson) The diameter of the hollow core of a solid propellant grain remains axially uniform during burning. Its length L is 5 m, and initially its inner diameter d is 0.37 m; the outer diameter of the grain D is 0.82 m. The figure below explains the geometry.



The burning rate r_b is 1.2 cm/s, which as a first approximation may be assumed to be uniform over the entire inner surface of the grain. The grain density is 1875 kg/m^3. The combustion chamber stagnation pressure and stagnation temperature downstream of the grain (at (2)) are 2.17 MPa and 2580K, respectively. The gas specific ratio and molecular weight are 1.2 and 20, respectively.

Neglecting the effects of friction, but recognizing that there is a nonzero flow of gas at the end of the grain, (2), estimate **at the beginning and end of combustion**,

a.) The Mach number M_2 at the downstream end of the grain.

b.) The static pressure ratio p_2/p_1 along the length of the grain.