16.50 Propulsion Systems
Optional Heat Transfer Homework

Consider a rocket with a combustor-nozzle contour specified through a Mach number distribution as follows:

$$M = 0.2 + 17.2667\left(\frac{x}{L}\right)^2 - 19.3333\left(\frac{x}{L}\right)^3 + 5.8667\left(\frac{x}{L}\right)^4$$

which is arranged so as to place a sonic point at x/L=0.25 and to reach M=4 at the exit (x=L). x is measured from the injector plane.

(a) Assuming γ = 1.2, generate and plot profiles of A/A_t, P/P_c and T/T_c versus x/L.

As additional data, take a molecular mass of 20 g/mole, $P_c = 70\ atm$, $T_c = 3300\ K$, $T_w = 1{,}000\ K$ and $L/D_e = 2$. The gas viscosity is approximately given by $\mu_g = 3 \times 10^{-5}(T/T_c)^{0.6}$. The absolute size of the rocket is determined by the requirement to generate a matched thrust $F_{match} = 2 \times 10^6 N$.

(b) Calculate and plot the wall heat flux distribution. Multiply times the approximate wall area $\pi D(x)dx$ of a nozzle element of length dx and integrate to find the total amount Q of heat loss (use a simple trapezoidal rule integration). Calculate $Q/(\dot{m}c_pT_c)$, the fraction of input power that is lost to the wall.

(c) Assume cooling is done regeneratively by the kerosene fuel, which enters the cooling passages at 300K, and calculate the temperature reached by this fuel as it leaves the cooling passages to be injected into the combustor. Take the Oxidizer/Fuel mass ratio as $O/F = 2.4$, and assume the specific heat of the liquid kerosene is 2,090 J/kg/K.