Equations

Compressible Flow Equations for Combustion Chamber (downstream) & Nozzle:

$$\frac{P}{P_0} = [1 + \frac{k-1}{2}Ma^2]^{\frac{-k}{k-1}} \tag{1}$$

$$\frac{\rho}{\rho_0} = [1 + \frac{k-1}{2}Ma^2]^{\frac{-1}{k-1}} \tag{2}$$

$$\frac{T}{T_0} = [1 + \frac{k-1}{2}Ma^2]^{-1} \tag{3}$$

$$\frac{A}{A^*} = \frac{1}{Ma}[(\frac{2}{k+1})(1 + \frac{k-1}{2}Ma^2)]^{\frac{k+1}{2(k-1)}} \tag{4}$$

Nozzle Equations:

$$\vec{T} = A^* P_0 \sqrt{\frac{2k^2}{k-1}(\frac{2}{k+1})^{\frac{k+1}{k-1}}[1 - (\frac{P_e}{P_0})^{\frac{k-1}{k}}]} + (P_e - P_a)A_e \tag{5}$$

$$A_e = \frac{\dot{w}V_e}{v_e} = \frac{\vec{T} g_0 T_c R}{P_c MW (v_e)^2}(\frac{P_c}{P_e})^{\frac{1}{k}} \tag{6}$$

$$c = v_e \zeta_v \tag{7}$$

$$v_e = \bar{v_e} \zeta_v \tag{8}$$

$$I_{sp} = \frac{\bar{v_e}}{g_0} \tag{9}$$

Combustion Chamber Equations:

$$D_c = \sqrt{\frac{4A_c}{\pi}} \tag{10}$$

$$L_c = \frac{V_c}{A_c} \tag{11}$$

$$L^* = \frac{V_c}{A_t} \tag{12}$$

$$\sigma_1 = \frac{P_c D_c}{2t} \tag{13}$$

$$\sigma_2 = \frac{P_c D_c}{4t} \tag{14}$$

$$\sigma_T = E\alpha\Delta T \tag{15}$$

Injector Equations:

$$\dot{w} = \dot{m}g_0 = \frac{\vec{T} g_0}{v_e} \tag{16}$$

$$\dot{w}_o = \frac{(\dot{w})(MR)}{MR+1} \tag{17}$$

$$\dot{w}_f = \frac{\dot{w}}{MR+1} \tag{18}$$

$$\dot{Q}_o = \frac{\dot{w}_o}{\rho_o} \tag{19}$$

$$\dot{Q}_f = \frac{\dot{w}_f}{\rho_f} \tag{20}$$

$$\Sigma A_o = \frac{\dot{w}_o}{C_d\sqrt{2g_0\Delta P\rho_o}} \tag{21}$$

$$\Sigma A_f = \frac{\dot{w}_f}{C_d\sqrt{2g_0\Delta P\rho_f}} \tag{22}$$

$$v_o = C_d\sqrt{2g_0\frac{\Delta P}{\rho_o}} \tag{23}$$

$$v_f = C_d\sqrt{2g_0\frac{\Delta P}{\rho_f}} \tag{24}$$

$$h_L = h_{L,major} + h_{L,minor} \tag{25}$$

$$h_{L,minor} = K_L\frac{v^2}{g_0} \tag{26}$$

Thermodynamic/Heat Transfer Equations:

$$q_{TOTAL} = hA(T - T_\infty) = \frac{\Delta T}{R_{TOTAL}} \tag{27}$$

$$R_{COND} = \frac{ln(r_2/r_1)}{2k\pi L} \tag{28}$$

$$R_{CONV} = \frac{1}{hA} \tag{29}$$

$$\frac{T_{aw} - T_{wg}}{T_{aw} - T_{co}} = exp(-\frac{h_g}{G_c c_{pvc}\eta_c}) \tag{30}$$

Constants

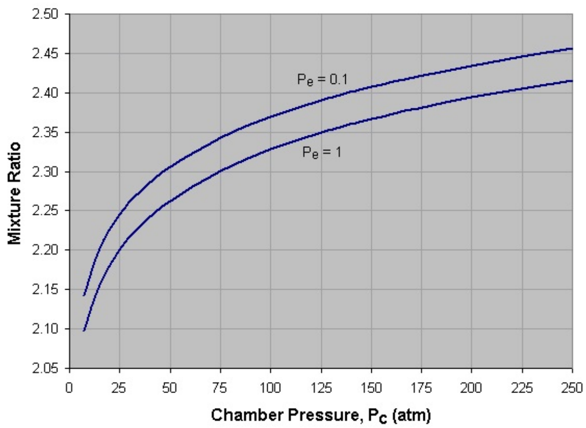
- **Specific Heat Ratio:** $k = 1.2$
- **Atmospheric Pressure:** $P_a = 101325\,\text{Pa} = 14.7\,\text{PSI}$
- **Universal Gas Constant:** $R = 1544\,\frac{\text{lb*ft}}{\text{lb*mol*}^\circ\text{R}}$

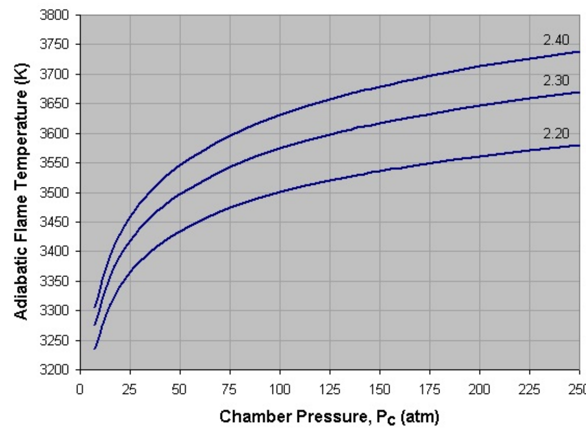
Material Properties

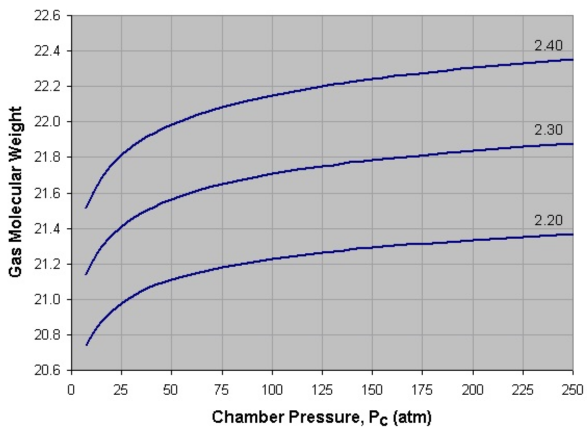
Property	**Units**	**Aluminum**	**Graphite**	**Steel**	**Alumina**	**Carbon Fiber**
Type	–	6061-T6	C-580 Superfine	SAE 1020	UT391 Fiber Cloth	2×2 twill
Density (ρ)	lb/in^3	0.0975	0.064	0.284	–	–
Yield Strength (σ_{YS})	PSI	40000	–	42000 (85F)	250000	–
Elastic Modulus (E)	KSI	10000	$1160 - 2175$	29500	22000	–
Specific Heat Capacity (c_p)	$BTU/lb^\circ F$	0.214	$0.169 - 0.198$	0.115	–	–
Thermal Conductivity (k)	$BTUin/ft^2hr^\circ F$	1160	$173 - 3258$	350.1	1.6	–
Thermal Expansion (α)	$\mu in/\mu in^\circ F$	14	1.1	6.5	–	–
Melting Point (T_{MP})	$^\circ F$	$1080 - 1205$	7820	2780	3300	–

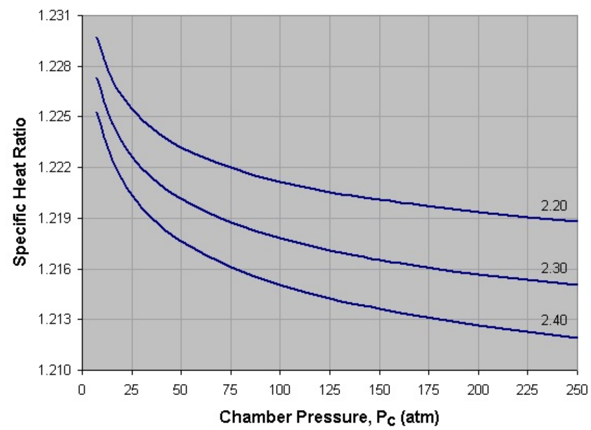
Emperical Charts

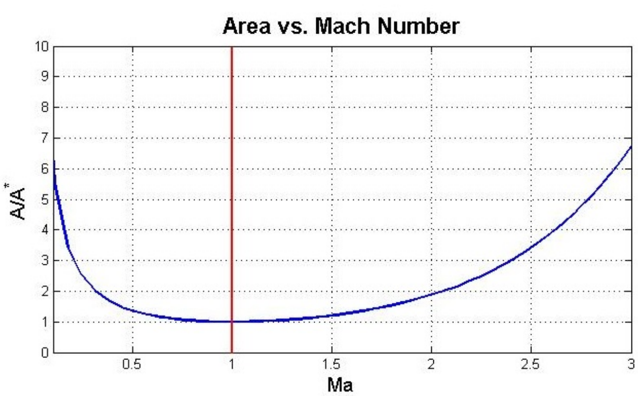
RP1 & LOX Optimum Mixture Ratio, Adiabatic Flame Temperature, Exhaust Molecular Weight & Specific Heat Ratio (Top) & Compressible Flow Solutions for k=1.2 (Bottom):

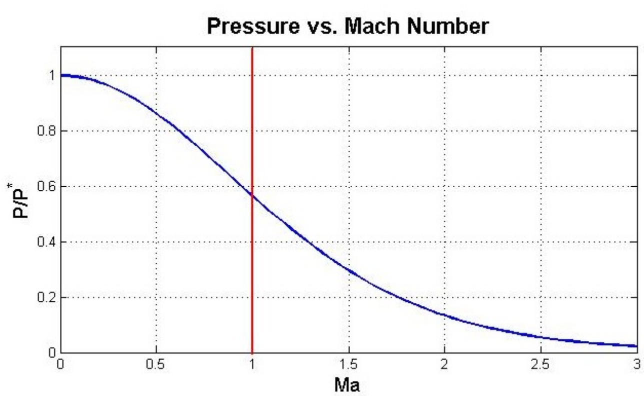


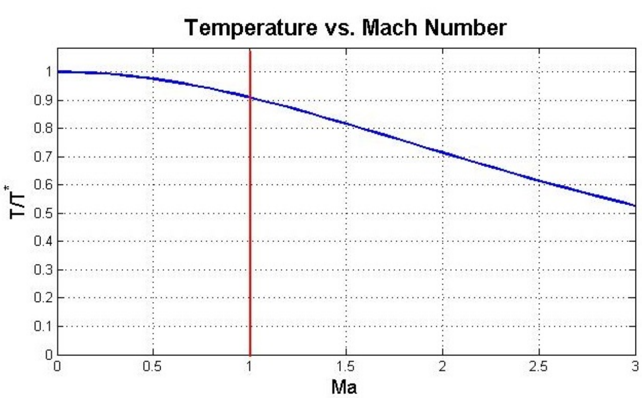


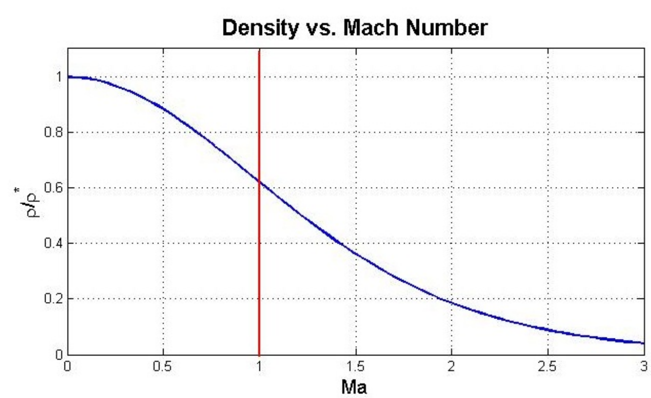












Engine Design Parameters

Below are the performance parameters chosen for our engine design. Values from the Emperical Charts are used here:

- **Propellants:** Kerosene ($MW_{FUEL} = 170.34\,\frac{\text{g}}{\text{mol}}$) & Liquid Oxygen ($MW_{OXIDIZER} = 32.00\,\frac{\text{g}}{\text{mol}}$)
- **Thrust:** $\vec{T} = 4448.22\,\text{N} = 1000\,\text{lbf}$
- **Mixture Ratio:** $MR = \frac{(n_O)(MW_{OXIDIZER})}{(n_F)(MW_{FUEL})} = 2.348$
- **Chamber Pressure:** $P_c = 2018120\,\text{Pa} = 292.7036\,\text{PSI}$
- **Chamber Temperature:** $T_c = 3425\,\text{K} = 6165°\,\text{R}$
- **Stoichiometry:** $C_{12}H_{26} + 12.5O_2 \rightarrow 12CO + 13H_2O$
- **Exhaust Molecular Weight:** $MW = 21.6\,\frac{\text{lb}}{\text{mol}}$

Thermo-Chemical Analysis

USC Flame Temperature Calculator:

12.500000 moles of O_2 at temperature 100.000000 K
1.000000 moles of $C_{12}H_{26}$ at temperature 298.000000 K
Reactant Pressure: 19.910000 atm
Flame temperature: 3451.774170 K
Mole fractions of products (those below 1E-6 are not listed):

H: 0.042702	H_2O: 0.311889	OH: 0.046399	CO: 0.346481	C_2H_4: 0.000014
HO_2: 0.000042	H_2O_2: 0.000004	O_2: 0.008384	CO_2: 0.103828	
H_2: 0.131379	O: 0.008876	HCO: 0.000015	$C_8H_{18}(L)$: 0.000024	

Average Exhaust Molecular Weight: 21.395788 $\frac{\text{lb}}{\text{mol}}$

Error in Chamber Temperature: $\epsilon_T = \frac{3451.77417-3425}{3425} = 0.7817\%$
Error in Exhaust Molecular Weight: $\epsilon_{MW} = \frac{21.6-21.395788}{21.6} = 0.9454\%$

Nozzle

Assumptions:

Isentropic flow throughout the nozzle, Choked flow at the throat ($Ma_t = 1$) to satisfy supersonic solution, Chamber pressure is the same as total pressure ($P_c = P_0$), and Chamber temperature is the same as total temperature ($T_c = T_0$).

Specifications:

Parameter	Inlet	Throat	Exit
Radius	$r_i = 0.052337\,\text{m}$	$r_t = 0.022325\,\text{m}$	$r_e = 0.043745\,\text{m}$
Area	$A_i = 0.00860523\,\text{m}^2$	$A_t = 0.00156579\,\text{m}^2$	$A_e = 0.00601183\,\text{m}^2$
Mach Number	$Ma_i = 0.108$	$Ma_t = 1$	$Ma_e = 2.589$
Pressure	$P_i = 2018120\,\text{Pa}$	$P_t = 1139176.1\,\text{Pa}$	$P_e = 101325\,\text{Pa}$
Temperature	$T_i = 3425\,\text{K}$	$T_t = 3113.63\,\text{K}$	$T_e = 2050.54\,\text{K}$

- **Nozzle Length:** $L_n = 0.06433\,\text{m} = 2.53268\,\text{in}$
- **Converging Half Angle:** $\theta = 45°$
- **Diverging Half Angle:** $\alpha = 15.906°$
- **Thrust Coefficient:** $C_T = \frac{\vec{T}}{A_t P_c} = \frac{1000}{(2.42698)(292.7036)} = 1.40768$
- **Theoretical Exhaust Velocity:** $v_e = \sqrt{\frac{\vec{T} g_0 T_c R}{A_e P_c MW}\left(\frac{P_c}{P_e}\right)^{\frac{1}{k}}} = \sqrt{\frac{(1000)(32.2)(6165)(1544)}{(9.318355)(292.70356)(21.6)}(19.9118)^{\frac{1}{1.2}}} = 7932.38\,\frac{\text{ft}}{\text{s}}$

  v_e is also noted as C^*

  Theoretical Exhaust Velocity relies on correction factor of $\zeta_v = 0.94$ (Sugessted by Sutton)

  Actual Effective Exhaust Velocity: $\bar{v_e} = v_e\zeta_v = 7456.4\,\frac{\text{ft}}{\text{s}}$
- **Specific Impulse:** $I_{sp} = \frac{\bar{v_e}}{g_0} = 231.57\,\text{s}$

Chamber Configuration

Assumptions:

"The customary method of establishing the L* of a new thrust chamber design largely relies on past experience with similar propellants and engine size. Under a given set of operating conditions, such as type of propellant, mixture ratio, chamber pressure, injector design, and chamber geometry, the value of the minimum required L* can only be evaluated by actual firings of experimental thrust chambers. Typical L* values for various propellants are shown in the table below." (Braeunig)

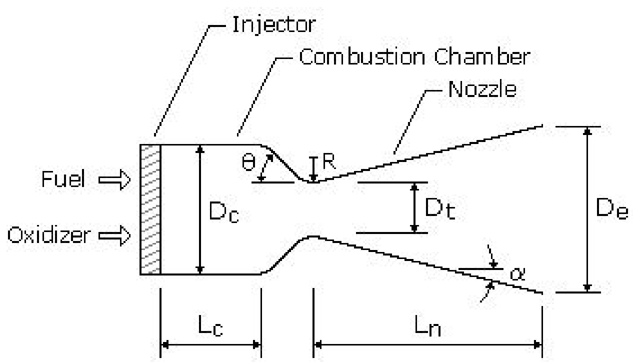


Figure 1.4

Propellant Combination	L*, cm
Nitric acid/hydrazine-base fuel	76-89
Nitrogen tetroxide/hydrazine-base fuel	76-89
Hydrogen peroxide/RP-1 (including catalyst bed)	152-178
Liquid oxygen/RP-1	102-127
Liquid oxygen/ammonia	76-102
Liquid oxygen/liquid hydrogen (GH_2 injection)	56-71
Liquid oxygen/liquid hydrogen (LH_2 injection)	76-102
Liquid fluorine/liquid hydrogen (GH_2 injection)	56-66
Liquid fluorine/liquid hydrogen (LH_2 injection)	64-76
Liquid fluorine/hydrazine	61-71
Chlorine trifluoride/hydrazine-base fuel	51-89

Specifications:

- **Cross Sectional Chamber Area:** $A_c = 14.0553\,\text{in}^2$

  Cross sectional area of the chamber is evaluated using the outer radius of the nozzle ($r_o = 2.1152\,\text{in}$).

- **Cross Sectional Chamber Diameter:** $D_c = \sqrt{\frac{4A_c}{\pi}} = 4.23034\,\text{in}$

- **Critical Combustion Chamber Wall Thickness using Hoop Stress:** $\sigma_1 = \frac{P_c D_c}{2t_w} \rightarrow t_w = \frac{P_c D_c}{2\sigma_1}$

  Typical Factor of Safety range for Pressure Vessels: $3.5 \leq FS \leq 6$

  Maximum allowable stress is Yield Stress therefore: $\sigma_1 = \sigma_{YS}$

  Critical Aluminum Chamber Wall Thickness ($FS = 3.5$): $t_w = \frac{P_c D_c}{2(FS)(\sigma_{YS})} = \frac{(292.7036)(2.23034)}{2(3.5)(40000)} = 0.002331\,\text{in}$

  Critical Steel Chamber Wall Thickness ($FS = 3.5$): $t_w = \frac{P_c D_c}{2(FS)(\sigma_{YS})} = \frac{(292.7036)(2.23034)}{2(3.5)(42000)} = 0.002221\,\text{in}$

- **Conservative Characteristic Chamber Length:** $L^* = 105\,\text{cm} = 41.3386\,\text{in}$

  Shorter length to avoid downstream recombination

  Combustion Chamber Volume: $V_c = L^* A_t = 100.3279\,\text{in}^3$

  Combustion Chamber Length: $L_c = \frac{V_c}{A_c} = 7.1381\,\text{in}$

- **Averaged Characteristic Chamber Length:** $L^* = 114\,\text{cm} = 44.8819\,\text{in}$

  Average value of characteristic length range

  Combustion Chamber Volume: $V_c = L^* A_t = 108.9274\,\text{in}^3$

  Combustion Chamber Length: $L_c = \frac{V_c}{A_c} = 7.7499\,\text{in}$

Injector Design

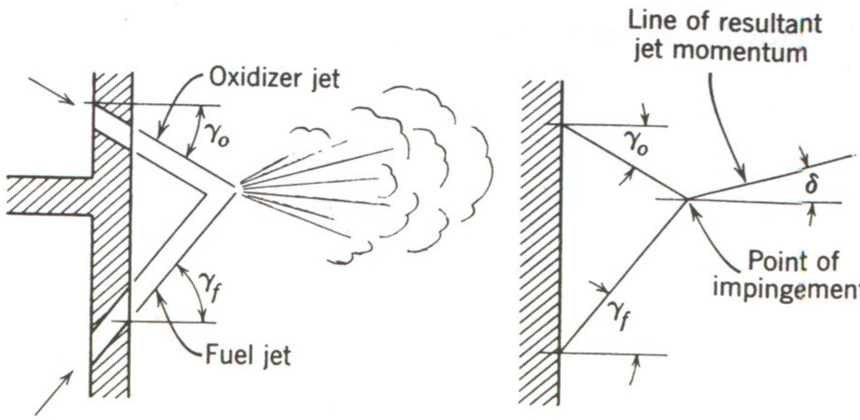
Assumptions:

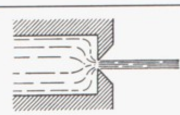
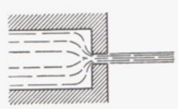
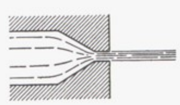
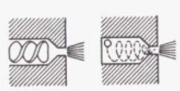
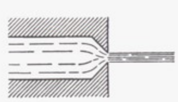
"It is not uncommon for an injector with C* efficiency as low as 92% to be considered acceptable. High levels of combustion efficiency derive from uniform distribution of the desired mixture ratio and fine atomization of the liquid propellants. Local mixing within the injection-element spray pattern must take place at virtually a microscopic level to ensure combustion efficiencies approaching 100%." (Braeunig)

Specific Gravity of a substance is the ratio of density of the substance to the density of water at $4°C$.

Typical ΔP is $15 - 25\%$ of P_c. "It is assumed that the injection pressure drops in the fuel and oxidizer lines are equal to — PSI and that both orifice discharge coefficients are equal to —." (Sutton)

"The injection angles are chosen so that the resultant momentum will be in an axial direction, inclination of the oxidizer jet is arbitrarily chosen." (Sutton)



Orifice Type	Diagram	Diameter (mm)	Discharge Coefficient
Sharp-edged orifice		Above 2.5 Below 2.5	0.61 0.65 approx.
Short-tube with rounded entrance $L/D > 3.0$		1.00 1.57 1.00 (with $L/D \sim 1.0$)	0.88 0.90 0.70
Short tube with conical entrance		0.50 1.00 1.57 2.54 3.18	0.7 0.82 0.76 0.84–0.80 0.84–0.78
Short tube with spiral effect		1.0–6.4	0.2–0.55
Sharp-edged cone		1.00 1.57	0.70–0.69 0.72

Fuel Properties:

Table 3 Typical aviation fuel properties[15,17,20,35,44–46]

Property	Avgas	JP-4	JP-5	JP-7	JP-8 (Jet A/A-1)	RP-1
Approximate formula[a]	C_7H_{15}	$C_{8.5}H_{17}$	$C_{12}H_{22}$	$C_{12}H_{25}$	$C_{11}H_{21}$	$C_{12}H_{23}$
H/C ratio	2.09	2.00	1.92	2.07	1.91	1.95
Boiling range, °F (°C)	115–295 (46–145)	140–460 (60–240)	360–495 (180–260)	370–480 (190–250)	330–510 (165–265)	350–525 (175–275)
Freeze point, °F (°C)[b]		−80 (−62)	−57 (−49)	−47 (−44)	JP-8/Jet A-1: −60 (−51); Jet A: −50 (−45)	−55 (−48)
Flash point, °F (°C)		−10 (−23)	147 (64)	140 (60)	127 (53)	134 (57)
Net heating value, Btu/lb (kJ/kg)		18,700 (43,490)	18,500 (43,025)	18,875 (43,895)	18,550 (43,140)	18,650 (43,370)
Specific gravity, 16°C (60°F)	0.72	0.76	0.81	0.79	0.81	0.81
Critical temperature, °F (°C)		620 (325)	750 (400)	750 (400)	770 (410)	770 (410)
Critical pressure, psia (atm)		450 (30.5)	290 (19.5)	305 (20.5)	340 (23)	315 (21.5)
Average composition						
Aromatics, vol%	25	10	19	3	18	3
Naphthenes		29	34	32	35	58
Paraffins		59	45	65	45	39
Olefins	10	2	2		2	
Sulfur, ppm		370	470	2	490	20

[a]For illustration of average carbon number, not designed to give accurate H/C ratios.
[b]Typical.

- Kerosene is a crude-oil distillate similar to petrodiesel but with a wider-fraction distillation. Jet fuel is kerosene-based, with special additives ($< 1\%$). Rocket propellant RP-1 (also named Refined Petroleum) is a refined jet fuel, free of sulfur and with shorter and branched carbon-chains more resistant to thermal breakdown.
- Jet-A a narrow cut kerosene product. This is the standard commercial and general aviation grade available in the United States. It usually contains no additives but may be additized with a anti-icing chemicals.
- Jet-A1 is identical to Jet-A with the exception of freeze-point. Used outside the US and is the fuel of choice for long haul flights where the fuel temperature may fall to near the freeze point. Often contains a static dissipator additive.
- Jet-B is a wide cut kerosene with lighter gasoline type naphtha components. Used widely in Canada, contains static dissipator and has a very low flash point.
- JP-4 is a military designation for a fuel like Jet-B but contains a full additive package including corrosion inhibitor, anti-icing and static dissipater. JP-4 is typically composed of about 50-60% gasoline and 40-50% kerosene, is highly volatile, and contains hydrocarbons in the C4-C16 range. Compare to Jet-B.
- JP-5 is another military fuel. It has a higher flash point (140 F min.) and was designed for use by the US Navy on board aircraft carriers. It contains anti-ice and corrosion inhibitors.
- JP-8 is like Jet-A1 with a full additive package.

Oxidizer Properties:

Property	Value
Molecular Weight	$MW = 32.00\,\frac{\text{g}}{\text{mol}}$
Boiling Point @ 1 atm	$T_{BP} = -297.4^{\circ}\,\text{F}$
Freezing Point @ 1 atm	$T_F = -361.9^{\circ}\,\text{F}$
Critical Temperature	$T_{CR} = -181.8^{\circ}\,\text{F}$
Critical Pressure	$P_{CR} = 729.1\,\text{PSI}$
Density, Liquid @ BP	$\rho_l = 71.23\,\frac{\text{lb}}{\text{ft}^3}$
Density, Gas @ $68^{\circ}F$	$\rho_g = 0.0831\,\frac{\text{lb}}{\text{ft}^3}$
Specific Gravity, Gas (air=1) @ $68^{\circ}F$	$SG = 1.11$
Specific Gravity, Gas (water=1) @ $68^{\circ}F$	$SG = 1.14$
Specific Volume @ $68^{\circ}F$, 1 atm	$v = 12.08\,\frac{\text{lb}}{\text{ft}^3}$

Specifications:

Propellant	RP-1	LOX
Temperature of Injected Propellant	298 K	100 K
Specific Gravity at Injection Temperature	0.820	1.14
Density at Injection Temperature	$51.197\,\text{lb/ft}^3$	$71.2303\,\text{lb/ft}^3$
Heat of Vaporization at STP (BTU/lb)	108	91.7
Boiling Point at 1 atm	$302 - 572^{\circ}\,\text{F}$	$-297.4^{\circ}\,\text{F}$
Boiling Point at 292.7036 PSI	–	–

- 20% Injection Drop is assumed therefore: $\Delta P = 0.2P_c = 58.5407\,\text{PSI}$
- Number of injection streams pairs are arbitrarily chosen: $n = 16$
- Orifice type: Short Tube with Rounded Entrance (simplest design).

  Sutton assumes $C_d = 0.75$ for both orifices.
- Design constraint of 1 inch impingement point.
- **Propellant Weight Flow:** $\dot{w} = \dot{m}g_0 = \frac{\vec{T}g_0}{v_e} = \frac{(1000)(32.2)}{7932.38} = 4.05931\,\frac{\text{lb}}{\text{s}}$
- **Oxidizer Weight Flow:** $\dot{w}_o = \frac{(\dot{w})(MR)}{MR+1} = \frac{(4.05931)(2.348)}{2.348+1} = 2.8468\,\frac{\text{lb}}{\text{s}}$
- **Fuel Weight Flow:** $\dot{w}_f = \frac{\dot{w}}{MR+1} = \frac{4.05931}{2.348+1} = 1.21245\,\frac{\text{lb}}{\text{s}}$
- **Oxidizer Volumetric Flow Rate:** $\dot{Q}_o = \frac{\dot{w}_o}{\rho_o} = 2.8468 \times \frac{1728}{71.2303} = 69.06149\,\frac{\text{in}^3}{\text{s}}$
- **Fuel Volumetric Flow Rate:** $\dot{Q}_f = \frac{\dot{w}_f}{\rho_f} = 1.21245 \times \frac{1728}{51.197} = 40.9225\,\frac{\text{in}^3}{\text{s}}$
- **Total Oxidizer Injection Area:** $\Sigma A_o = \frac{\dot{w}_o}{C_d\sqrt{2g_0\Delta P\rho_o}} = \frac{2.8468\times 144}{0.75\sqrt{(2)(32.2)(58.5407)(144)(71.2303)}} = 0.087897\,\text{in}^2$

  Oxidizer Hole Area: $A_o = \frac{\Sigma A_o}{n} = 0.0054935\,\text{in}^2$
- **Total Fuel Injection Area:** $\Sigma A_f = \frac{\dot{w}_f}{C_d\sqrt{2g_0\Delta P\rho_f}} = \frac{1.21245\times 144}{0.75\sqrt{(2)(32.2)(58.5407)(144)(51.197)}} = 0.044156\,\text{in}^2$

  Fuel Hole Area: $A_f = \frac{\Sigma A_f}{n} = 0.0027598\,\text{in}^2$
- **Oxidizer Injection Velocity:** $v_o = C_d\sqrt{2g_0\frac{\Delta P}{\rho_o}} = 0.75\sqrt{(2)(32.2)\frac{58.5407\times 144}{71.2303}} = 65.476\,\frac{\text{ft}}{\text{s}}$
- **Fuel Injection Velocity:** $v_f = C_d\sqrt{2g_0\frac{\Delta P}{\rho_f}} = 0.75\sqrt{(2)(32.2)\frac{58.5407\times 144}{51.197}} = 77.231\,\frac{\text{ft}}{\text{s}}$
- **Fuel Injection Angle:** $sin\gamma_f = MR(\frac{v_o}{v_f})sin\gamma_o = 2.348(\frac{65.476}{77.231})sin(20^{\circ}) = 0.68083$

  For $\gamma_o = 20^{\circ}$, $\gamma_f = 42.9087^{\circ}$

Heat Transfer Analysis

Assumptions:

The ablative mold will be comprised of layers of Alumina and Carbon Fibers impregnated with a phenolic or epoxy resin. Ablative mold will be created by tape-wrapping on a shaped mandrel. "Ablation process is a combination of surface melting, sublimation, charring, evaporation, decomposition in depth, and film cooling" (Sutton).

"It is desirable to reduce both the thickness of film cooling layer and the mass flow of cooler gas, relative to the total flow, to a practical minimum" (Sutton).

Proposed engine configuration is acceptable if assuming steady state ($\dot{q} = 0$); Our projected burn time is roughly 3 seconds, We can neglect q_{RAD} because emissivity of ablative material (ϵ_A) is low.

We need to determine the inner (T_i) and outer (T_o) temperatures of the combustion chamber but they depend on the total heat transfer (q_{TOTAL}) which depends on the reference combustion chamber temperature of hot gases (T_g), therefore we can make two assumptions:

1. $T_g = T_c = 3425\,\mathrm{K}$ (Using adiabatic flame temperature may lead to over-design)
2. $T_g < T_c$ (Assuming $3000\,\mathrm{K} \le T_g < 3425\,\mathrm{K}$)

Specifications:

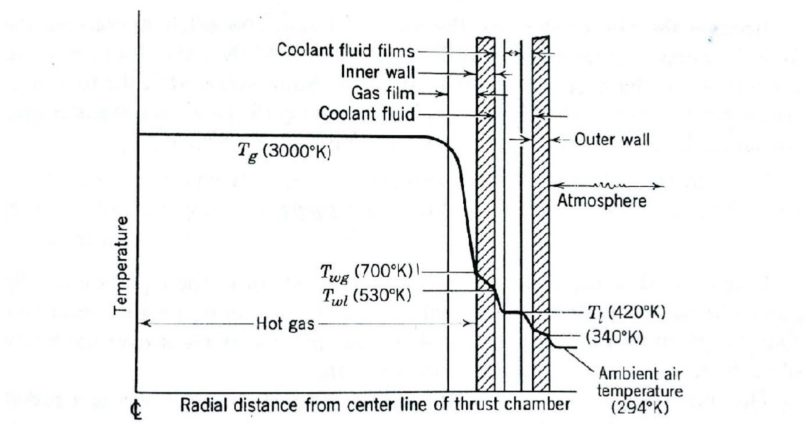
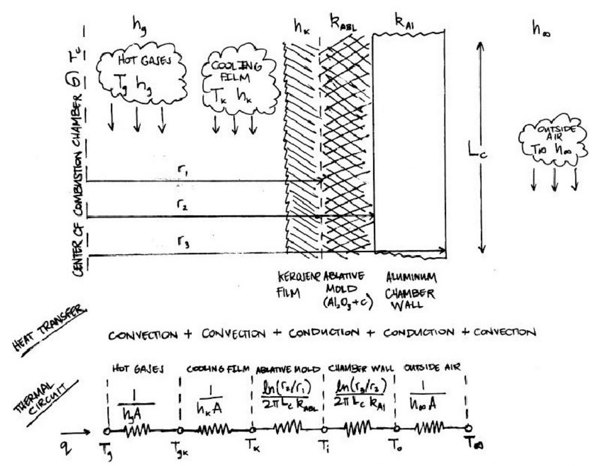


Fig. 4–1. *Temperature gradients in cooled rocket thrust chamber. Given temperature values are typical.*



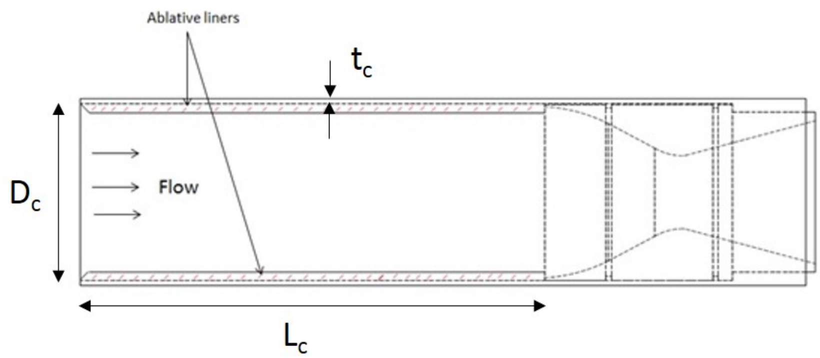
T_g	h_g	h_k	k_{ABL}	k_{Al}	h_∞	T_∞
3200 K	–	–	$269.57 - 864.88\,\frac{\mathrm{BTUin}}{\mathrm{ft^2hr^\circ F}}$	$1160\,\frac{\mathrm{BTUin}}{\mathrm{ft^2hr^\circ F}}$	–	298 K

- Assuming $T_g < T_c$ such that $T_g = 3200\,\mathrm{K}$
- **Chamber Melting Temperature**: $T_i = 1080^\circ\,\mathrm{F} = 855.372\,\mathrm{K}$

  Lower melting temperature chosen for design process
- **Total Thermal Resistance:** $\Sigma R_T = \frac{1}{Ah_g} + \frac{1}{A_1 h_k} + \frac{ln(r_2/r_1)}{2k_A L\pi} + \frac{ln(r_3/r_2)}{2k_{Al} L\pi} + \frac{1}{A_3 h_\infty}$

Proposed Design

Engine Assembly:



- **Cross Sectional Chamber Diameter:** $D_c = 4.23034\,\text{in}$
- **Combustion Chamber Volume:** $V_c = 108.9274\,\text{in}^3$
- **Combustion Chamber Characteristic Length:** $L_c = 7.7499\,\text{in}$
- **Combustion Chamber Thickness (for $L_c = 7.7499\,\text{in}$):** $t_c = 0.128937\,\text{in}$
- **Combustion Chamber Notch Thickness (for Nozzle):** $t_n = 0.325\,\text{in}$
- **Ablative Liner Thickness:** $t_a =???\,\text{in}$
- **Consumed Fuel Volume for 3 second burn:** $V_F = 2.01168\,\text{L}$
- **Consumed Oxidizer Volume for 3 second burn:** $V_O = 3.39507\,\text{L}$

Actual Aluminum Chamber Dimensions:

- **Cross Sectional Chamber Diameter:** $D_{outer} = 4.5\,\text{in}$
- **Chamber Thickness:** $t_c = 0.337\,\text{in}$
- **Chamber Length:** $L_c = 12\,\text{in}$

Bill of Materials:

Part	Dimensions	Quantity	Manufacturer	Part ID	Cost
6061-T6 Al Pipe	18.00” × 4.5” × 0.337”	1	Metals Depot	T3480	\$77.18
Ceramic Fiber	–	–	–	–	–
Carbon Fiber	–	–	–	–	–
Epoxy	–	–	–	–	–
Flared Tube Fittings	AN8	5	McMaster-Carr	2227K13	\$35.75
Flared Tube Fitting Plugs	AN8	3	McMaster-Carr	2227K53	\$11.67
Neoprene O-Rings	$D_o \times D_i = 0.625” \times 0.5”$	100	McMaster-Carr	94115K014	\$2.85
Jet Fuel (JP-4)	\$3.84/*gal*	2.01168 L	Local Airport	$9130-00-256-8613$	\$2.04
Liquid Oxygen (LOX)	–	3.39507 L	–	–	–

References

[1] Sutton, George Paul. *Rocket Propulsion Elements: An Introduction to the Engineering of Rockets*. New York: Wiley, 1992. Print.