Rocket Engine Performance Report (200N)

Kelvin Aerospace Lab

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Abstract

This report presents the complete performance calculation workflow for a small liquid rocket engine using ethanol and gaseous oxygen (GOX). Calculations include propellant flow distribution, nozzle sizing, combustion chamber geometry, and injector orifice sizing. All results are based on standard rocket propulsion equations. All input parameters are derived from calculations performed by NASA Chemical Equilibrium Analysis (CEA).

Contents

1 Input Parameters

1.1 Base Engine Parameters

Table 1: Base Engine Design Parameters

Parameter	Symbol	Value
Thrust (Sea Level)	F	$200.0000\,\mathrm{N}$
Chamber pressure	P_c	$1\,800\,000.0000\,\mathrm{Pa}$
O/F mass ratio	O/F	1.60
Effective exhaust velocity	c	$2340.3250\,\mathrm{m/s}$
Characteristic velocity	c^*	$1734.0000\,\mathrm{m/s}$
Expansion ratio	$\epsilon = A_e/A_t$	3.8930
Standard gravity	g_0	$9.81\,\mathrm{m/s^2}$

2 Basic Performance Calculations

2.1 Specific Impulse

$$I_{sp} = \frac{c}{g_0} \tag{1}$$

$$I_{sp} = \frac{2340.3250\,\mathrm{m/s}}{9.81\,\mathrm{m/s^2}} = 238.565\,\mathrm{s} \tag{2}$$

2.2 Propellant Mass Flow Rates

Total mass flow rate:

$$\dot{m}_{\text{total}} = \frac{F}{c} = 0.0855\,\mathrm{kg/s} \tag{3}$$

Fuel and oxidizer flow rates (given O/F = OF):

$$\dot{m}_{\text{fuel}} = 0.0329\,\mathrm{kg/s} \tag{4}$$

$$\dot{m}_{\text{oxidizer}} = 0.0526\,\mathrm{kg/s} \tag{5}$$

3 Nozzle Design

3.1 Throat Area and Diameter

Throat area from characteristic velocity:

$$A_t = \frac{c^* \cdot \dot{m}_{\text{total}}}{P_c} \tag{6}$$

$$A_t = \frac{c^* \cdot \dot{m}_{\text{total}}}{P_c} = \frac{1734.0000\,\mathrm{m/s} \times 0.0855\,\mathrm{kg/s}}{1\,800\,000.0000\,\mathrm{Pa}} = 8.232\,475 \cdot 10^{-5}\,\mathrm{m^2}$$

$$d_t = 2\sqrt{\frac{A_t}{\pi}} = 2\sqrt{\frac{8.232\,475 \cdot 10^{-5}\,\mathrm{m^2}}{\pi}} \approx 10.24\,\mathrm{mm}$$

Throat diameter:

$$d_t = 2\sqrt{\frac{A_t}{\pi}} \approx 10.24\,\mathrm{mm} \tag{7}$$

3.2 Exit Area and Diameter

$$d_e = A_t \times \epsilon = 3.204\,902 \cdot 10^{-4}\,\mathrm{m}^2 \tag{8}$$

$$d_e = 2\sqrt{\frac{A_e}{\pi}} \approx 20.20\,\mathrm{mm} \tag{9}$$

4 Combustion Chamber Design

4.1 Characteristic Length

Selected value: $L^* = 1.100\,000\,\mathrm{m}$ (typical for small pressure-fed engines). Chamber volume:

$$V_c = L^* \times A_t = 1.100\,000\,\mathrm{m} \times 8.232\,475 \cdot 10^{-5}\,\mathrm{m}^2 = 0.0001\,\mathrm{m}^3 \approx 91\,\mathrm{cm}^3 \tag{10}$$

$$V_c = L^* \times A_t = 1.100\,000\,\mathrm{m} \times 8.232\,475 \cdot 10^{-5}\,\mathrm{m}^2 = 0.0001\,\mathrm{m}^3$$

4.2 Chamber Geometry for Different Contraction Ratios

Table 2: Combustion chamber dimensions for various A_c/A_t contraction ratios ($L^* = 1.100\,000\,\mathrm{m}$)

A_c/A_t	A_c (cm^2)	d_c (mm)	H_c (mm)	V_c (cm^3)
3	2.47	17.7	367	91
4	3.29	20.5	275	91
5	4.12	22.9	220	91
6	4.94	25.1	183	91
7	5.76	27.1	157	91
8	6.59	29.0	138	91
9	7.41	30.7	122	91
10	8.23	32.4	110	91
11	9.06	34.0	100	91
12	9.88	35.5	92	91
13	10.70	36.9	85	91
14	11.53	38.3	79	91
15	12.35	39.7	73	91
16	13.17	41.0	69	91
17	14.00	42.2	65	91
18	14.82	43.4	61	91
19	15.64	44.6	58	91
20	16.46	45.8	55	91
21	17.29	46.9	52	91
22	18.11	48.0	50	91

5 Chemical Reactions

The primary combustion reaction (stoichiometric with ethanol fuel) is:

$$C_2H_5OH + 3\,O_2 \rightarrow 2\,CO_2 + 3\,H_2O \tag{11}$$

Molar masses and thermodynamic properties used for calculations:

- Ethanol (liquid): molar mass $M_{\mathrm{ethanol}} = 46.070$ g/mol, injection density $\rho_{\mathrm{ethanol}} = 789.0000\,\mathrm{kg/m^3}$.
- Oxygen (gaseous $\mathrm{O_2}$): molar mass $M_{\mathrm{O2}} = 31.998$ g/mol, specific gas constant $R_{\mathrm{O2}} = 259.8000\,\mathrm{J/kgK}$.

For liquid propellants, density is a material property independent of pressure and temperature at typical injection conditions.

6 Injector Design

Propellant Density Determination

For gaseous propellants (GOX) at subsonic conditions, density is calculated from ideal gas law:

$$\rho_{\mathrm{gas}} = \frac{P_{\mathrm{inj}}}{R \cdot T_{\mathrm{inj}}} \tag{12}$$

For liquid propellants (ethanol), standard material density is used.

For liquid ethanol at 293 K, standard density is used: $\rho_{\mathrm{ethanol}} = 789.0000\,\mathrm{kg/m^3}$ (13)

$$\rho_{\mathrm{GOX}} = \frac{2\,340\,000.0000}{259.8000\,\mathrm{J/kgK} \cdot 29} \tag{14}$$

6.1 Ethanol Injector

Table 3: Ethanol Injector Input Parameters

Parameter	Symbol	Value
Injection temperature	T_{inj}	$293.1500\,\mathrm{K}$
Injection density	ρ_{inj}	$789.0000\,\mathrm{kg/m^3}$
Pressure drop	ΔP	$360\,000.0000\,\mathrm{Pa}$
Injection pressure	P_{inj}	$2\,160\,000.0000\,\mathrm{Pa}$
Discharge coefficient	C_d	0.60

Using incompressible (liquid) orifice formula: $A = \dot{m}/(C_d\sqrt{2\rho\Delta P})$

$$A_{o,\mathrm{total}} = \frac{\dot{m}_{\mathrm{fuel}}}{C_d\sqrt{2\rho\Delta P}} = \frac{0.0329\,\mathrm{kg/s}}{0.60\,\sqrt{2\,789.0000\,\mathrm{kg/m^3}\,360\,000.0000\,\mathrm{Pa}}} = 2.298\,394 \cdot 10^{-6}\,\mathrm{m}^2$$

$$A_{o,\text{total}} \approx 2.30\,\text{mm}^2 \quad \rightarrow \quad d_o = 2\sqrt{\frac{A_{o,\text{total}}}{\pi}} \approx 1.71\,\text{mm}$$

Equivalent single orifice diameter:

$$d_o = 2\sqrt{\frac{A_{o,\text{total}}}{\pi}} \approx 1.71\,\text{mm} \tag{15}$$

6.2 GOX Injector

Table 4: GOX Injector Input Parameters

Parameter	Symbol	Value
Injection temperature	T_{inj}	293.1500 K
Injection density	ρ_{inj}	30.7246 kg/m^3
Pressure drop	ΔP	540 000.0000 Pa
Injection pressure	P_{inj}	2 340 000.0000 Pa
Discharge coefficient	C_d	0.60
Specific heat ratio	γ	1.400
Y-factor (compressibility)	Y	0.932
Number of orifices	n	8

Y-factor (compressibility correction) calculation: For subsonic compressible flow, the Y-factor accounts for gas expansion through the orifice:

$$Y = 1 - \frac{0.41 \cdot \Delta P}{\gamma \cdot P_{\text{inj}}} \tag{16}$$

$$Y = 1 - \frac{0.41 \cdot 540\,000.0000\,\text{Pa}}{1.400 \cdot 2\,340\,000.0000\,\text{Pa}} = 0.932$$

Using Y-corrected subsonic formula: $A = \dot{m}/(Y \cdot C_d \cdot \sqrt{2 \cdot \rho \cdot \Delta P})$

$$A_{o,\text{total}} = \frac{\dot{m}_{\text{oxidizer}}}{Y \cdot C_d \cdot \sqrt{2 \cdot \rho \cdot \Delta P}} \tag{17}$$

$$\begin{aligned} A_{o,\text{total}} &= \frac{\dot{m}_{\text{oxidizer}}}{Y \cdot C_d \cdot \sqrt{2 \cdot \rho \cdot \Delta P}} \\ &= \frac{0.0526\,\text{kg/s}}{0.932 \cdot 0.60 \cdot \sqrt{2 \cdot 30.7246\,\text{kg/m}^3 \cdot 540\,000.0000\,\text{Pa}}} \\ &= 1.631\,862 \cdot 10^{-5}\,\text{m}^2 \end{aligned}$$

$$A_{o,\text{total}} \approx 16.32\,\text{mm}^2$$

For 8 orifices, per-orifice dimensions:

$$A_{o,\text{each}} \approx 2.04\,\text{mm}^2 \qquad d_{o,\text{each}} = 2\sqrt{\frac{A_{o,\text{each}}}{\pi}} \approx 1.61\,\text{mm} \tag{18}$$

7 Design Summary

- Thrust (Sea Level): 200.0000 N
- Specific impulse: 238.565 s
- O/F ratio: 1.60
- Total mass flow: 0.0855 kg/s
- Throat diameter: 10.24 mm
- Exit diameter: 20.20 mm
- L^*: 1.100 000 m