Development of a Cost-Effective 1.5kN Liquid-Fueled Rocket Propulsion System

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1 Nomenclature

Symbols				**Acronyms**		
ϵ	=	Expansion ratio		CEA	=	Chemical equilibrium with applications
γ	=	Ratio of specific heats				
ρ	=	Density	g/cm^3	$COTS$	=	Commercial off-the-shelf
C^*	=	Characteristic velocity	m/s	DAQ	=	Data acquisition & control
C_d	=	Discharge coefficient		FOD	=	Foreign object debris
C_v	=	Valve flow coefficient		$GLOW$	=	Gross lift-off weight
f_d	=	Friction factor		$P\&ID$	=	Plumbing and instrumentation diagram
F_t	=	Thrust	N			
g_0	=	Acceleration due to gravity	m/s^2	PT	=	Pressure transducer
I_{sp}	=	Specific impulse	s	TC	=	Thermocouple
O/F	=	Oxidizer-to-fuel ratio		SF	=	Safety factor
Q	=	Volumetric flow rate	L/s	VDC	=	Direct current voltage

2 Initial System Characterization

The following section outlines the characterization and design process for Aphlex 1B, our second-generation bi-propellant liquid rocket engine, set to fly on our 30 kg launch vehicle, Callisto 1. Project Caelus's unique position as a 501(c)(3) non-profit organization consisting entirely of high school students has laid the foundation for a design approach fully commited to cost-effectiveness, simplicity, and reliability, given certain mission constraints and objectives as discussed below.

2.1 Objectives

The Callisto 1 system is set to the following objectives:

1. Reach an altitude of 1500 m ($\approx$ 5000 ft).

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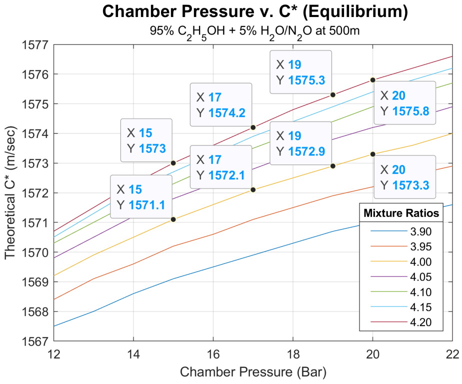

Figure 1: Theoretical C* efficiency vs chamber pressure and numerous mixture ratios.

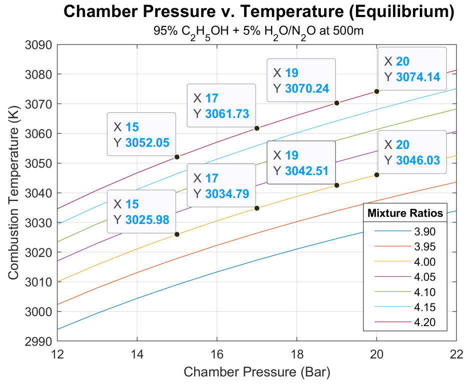

Figure 2: Combustion temperature vs chamber pressure and numerous mixture ratios.

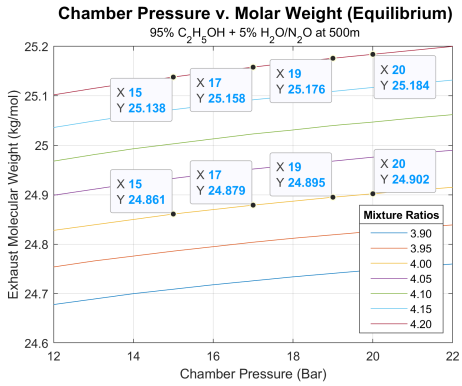

Figure 3: Gas molecular mass vs chamber pressure and numerous mixture ratios.

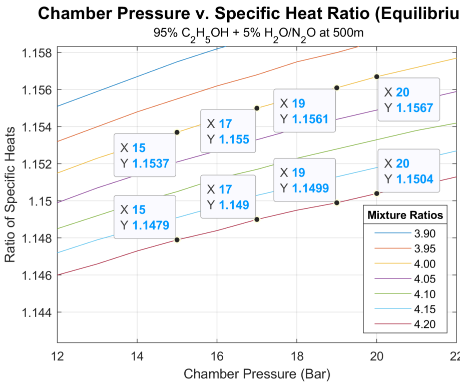

Figure 4: Ratio of specific heats vs chamber pressure and numerous mixture ratios.

2. A *GLOW* of no more than 30 kg ($\approx$ 70 lbsm).
3. A nominal main engine thrust of 1.5 kN ($\approx$ 350 lbsf).
4. A chamber pressure in the range of around 15 Bar to 20 Bar ($\approx$ 218 psi to 300 psi)
5. Consume a budget of no more than $10,000 USD.
6. Utililize 95% ethyl alcohol and nitrous oxide as the propellant combination.

2.2 Aphlex 1B

2.2.1 Chamber Pressure and Mixture Ratio Selection

The first step is to realize the theoretical maximum performance to be expected from our propellant combination. 95% ethyl alcohol (ethanol) was chosen for its availability, low pricing, and a modest specific impulse. A 95% dilution (by mass) with water was chosen as to lower the expected combustion chamber temperature. This tradeoff sacrifices some I_{sp} but reduces engineering complexity as regenerative and film cooling circuits may not be required. Industrial nitrous oxide was chosen as the main oxidizer for its self-pressurizing characteristics, non-cryogenic nature as opposed to liquid oxygen, relative ease to obtain, and a modest I_{sp} with ethanol. Using CEA, an open-source thermodynamics library provided by NASA's Glenn Research Center, critical data describing propellant combustion characteristics could be obtained. Numerical Python scripts were written to iterate through various combustion chamber pressures and mixture ratios and interface with CEA, and MATLAB scripts were used to parse and graph the CEA outputs as shown in Figures 1, 2, 3, and 4. Mixture ratios of 4.0 and 4.2 are labeled at certain chamber pressures. All dependent-variable properties (characteristic velocty C^*, combustion temperature T_c, exhaust molar mass M, and exhaust specific heat ratio γ) were permutated assuming shifting equilibrium flow and at an operational altitude of 500 m.

A mixture ratio of 4.0 was chosen for Aphlex 1B mainly in the interest of a conservative combustion temerature of around 3026 K and a middle-of-the-line theoretical C^* of around 1570 m/s, given a chamber pressure of 15 bar. This chamber pressure was chosen to minimize upstream fluid pressures and thus tank weight requirements, while also considering two additional design constraints that were not mentioned above but are nevertheless valid guiding parameters: a fluid flow velocity of less than 6 m/s to avoid water hammer effects [Zucrow Laboratories] and a pressure drop across the injector of around 25% of the designated chamber pressure [Michigan Aeronautical Science Association]. These guidelines further encourage a lower chamber pressure, and thus a conservative value of 15 Bar was chosen.

2.2.2 Nozzle Design

Design Parameters			
Name	Value	Unit	Uncertainty
Propellant (Fuel)	Ethanol (C_2H_5OH, 95%)	N/A	N/A
Propellant (Oxidizer)	Nitrous oxide (N_2O)	N/A	N/A
O/F, Oxidizer/fuel ratio	4.0	N/A	±1%
F_t, Nominal thrust	1.50	kN	±0.1%
P_c, Chamber static pressure	1.5×10^6	Pa	±0.1%
P_e, Ambient pressure	9.5540×10^4	Pa	±0.1%
T_c, Chamber static temperature	3025.98	K	±0.01%
M, Exhaust molecular mass	24.861	kg/mol	±0.01%
γ, Specific heat ratio	1.1537	N/A	±0.001%

Table 1: Summary of exhaust gas properties and fluid parameters.

$$T_3 = 15.04 - 0.00649h \tag{2.1}$$

$$P_3 = \left[101.29 \times \left(\frac{T + 273.1}{288.08}\right)^{5.256}\right] \times 1000 \tag{2.2}$$

The nozzle design process followed standard procedures outlined in Rocket Propulsion Elements [1] and open-source NASA documents. The thermodynamic properties of the exhaust gas and other important parameters are summarized and compiled in Table 1. The ambient pressure was calculated using NASA Glenn Research Center's Earth Atmosphere Model for an altitude within the troposphere (less than 11000 meters), as shown in Equations 2.1 and 2.2, where T_3 represents the ambient temperature in Kelvin, h is the altitude in meters, P_3 is the ambient pressure in Pascals.

Assuming isentropic flow in the supersonic nozzle with choked flow conditions at the throat, an ideal converging-diverging (de Laval) nozzle can be characterized.

2.3 Callisto 1

2.4 Plumbing System

The following section describes the theoretical process used to dimensionalize the prelimiary plumbing framework in preparation for the first static cold flow test. By definition, these calculations are purely speculatory and are only used for the initial design process. The purpose of the cold flow test is to verify these parameters, and consequently adjust these parameters to better fit the system requirements.

2.4.1 Assumptions

To simplify the rigorous analysis and optimization processes often associated with viscous pipe flow, a couple of assumptions are applied in the following section to both shorten the development timeline and to avoid unecessarily complex or expensive methods outside of the scope of a high school amateur rocketry program. They are as follows:

1. Flow is driven by both pressure and gravity.
2. Pipe is circular and is of constant cross-sectional area.

3. No swirl, circumferential variation, or entrance effects.
4. No shaft-work or heat-transfer effects.
5. Flow is fully developed (minimal boundary-layer effects).

Design Parameters			
Name	Value	Unit	Uncertainty
μ, Dynamic viscosity	0.00196	$kg/m \cdot s$	$\pm 0.1\%$
ρ, Density	786	kg/m^3	$\pm 1\%$
L, Pipe length	8.0	m	$\pm 0.1\%$
d, Pipe diameter	12.7	mm	$\pm 0.1\%$
$\dot{m}$, Mass flow rate	0.9133	kg/s	$\pm 0.1\%$
Pipe material	Hard nylon	N/A	N/A
ϵ, Roughness	1.5 to 40.0	μm	$\pm 50\%$

Table 2: Summary of exhaust gas properties and fluid parameters.

References

[1] George P. Sutton and Oscar Biblarz. *Rocket Propulsion Elements, 9th Edition*. Ninth. Wiley, Dec. 2016.