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Nitrous Oxide Ethanol Bi-Propellant Rocket Engine Design Process

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Abstract

An ethanol and nitrous oxide liquid bipropellant rocket engine was designed for static-fire testing. The two main priorities in the design of this engine were safety and ease of manufacture. The propellant feed system will be pressure-fed and use high-pressure nitrogen gas as a pressurant. Nitrogen will pressurize the ethanol and nitrous oxide run tanks to approximately 6.89Mpa each. With a mixture ratio of 3.18 and relatively high propellant densities, valves (such as pneumatically actuated ball valves) will be utilized in order to manage pressures, control propellant flow, and permit the system to vent when required. After pipe losses, nitrous oxide and ethanol are expected to be supplied to the injector at 6.49 and 6.34Mpa. Total propellant flow is expected to be 0.794Kg. These propellants were chosen due to their ease of procurement and handling. The nitrous oxide is expected to evaporate during injection. The chamber is expected to be maintained at 4.82Mpa and 2800K during firing. The engine will be approximately 30cm long. The combustion chamber is calculated to be 23cm long and 4cm wide, contracting to a throat slightly under 2cm wide. The nozzle will be conical and should be 5cm long and 4cm wide. Water will be pumped at approximately 1kg/s around the engine as external regenerative cooling. A thin copper wall will be used to conduct heat between the chamber and coolant. Other components will be made of stainless steel to ensure a similar thermal expansion coefficient. In addition, the avionics system will use an Arduino Mega 2560 R3 microcontroller to send and analyze data from the engine. The R3 will also interact with a DIGI XBee radio transceiver (at 2.4GHz, thus negating the need for a radio license). Finally, the engine's test stand will be constructed from 80/20 1010 series aluminum extrusions with the following dimensions: 1.5m x 0.4m x 0.4m. The purpose of this paper is to provide an in-depth analysis and explanation for the design of engines with similar size and capability.

1. Introduction

This project is being carried out by the Ryerson Propulsion Group (RPG) in order to design an ethanol and nitrous oxide liquid bipropellant rocket engine for static-fire testing that has dubbed this engine "Borealis". The primary objective of this project is to present a fully working prototype at the Launch Canada 2022 competition for a static-fire test, and a secondary goal is to provide students with a more hands-on approach to rocketry in order to ensure a successful

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transference of knowledge by demonstrating the complexities of rocket engine manufacturing and testing. It is important to note that the Borealis Engine is still in progress. The combustion chamber and cooling jacket will be manufactured using copper and stainless steel respectively. The emphasis of conventional machining techniques are central to the design philosophy, as limiting the complexity of the components allows for faster lead times and lower cost, while being equally effective. Furthermore, work continues on the propellant and electronics systems, a static test stand has been designed (and is under construction) for use with ethanol and nitrous oxide, which utilizes nitrogen gas as an inert pressurant. Alongside this are the control, telemetry, and communication systems needed to carry out and document operation. Finally, RPG's recently formed Joint Health & Safety Committee (JHSC) is to complete a risk assessment of not only the systems of the engine, but the eventual testing site location.

2. Methodology

2.1 Combustion System

The target chamber pressure of this design is 700psi (4.82Mpa) venting into sea level atmospheric pressure. The target flow rates are 0.19kg/s for ethanol and 0.604kg/s for N2O, with both propellants initially at room temperature. This data was inputted into NASA's Combustion Equilibrium Analysis software [1] to obtain the properties of the chamber gases.

Table 1: Exhaust Gas Properties

Avg Molar Mass	M	23.1 g/mol	Ratio of Specific Heats	γ	1.2
Avg Specific Heat	Cp	2000 Kj/Kg	Chamber Temperature	Tc	2800 K
Characteristic Velocity	C*	1535.8 m/s	Thrust Coefficient	Ct	1.52

C* and Ct assume perfect combustion efficiency. These values in reality would likely be 80% of these maximums. Using the above gas properties, calculations for the basic dimensions of the chamber and nozzle were done based on the guidelines put forth by Krzycki [2].

$$P_t = P_c\left[1 + \frac{\gamma-1}{2}\right]^{\frac{-\gamma}{\gamma-1}} \quad (1)$$

$$T_t = T_c\left[\frac{1}{1+\frac{\gamma-1}{2}}\right] \quad (2)$$

$$A_t = \frac{m_{total}}{P_t}\sqrt{\frac{8314\frac{Kj}{mol*K}*T_t}{M*\gamma}} \quad (3)$$

$$M_e = \sqrt{\frac{2}{\gamma-1}\left[\left(\frac{P_c}{P_{atm}}\right)^{\frac{\gamma-1}{\gamma}} - 1\right]} \quad (4)$$

$$A_e = \frac{A_t}{M_e}\left[\frac{1+\frac{\gamma-1}{2}M_e^{\ 2}}{\frac{\gamma+1}{2}}\right]^{\frac{\gamma+1}{2(\gamma-1)}} \quad (5)$$

The exit area ratio can be double-checked with CEA. The above energy conservation equation can also be used to determine the chamber diameter, choosing a mach number of 0.1 for the gases inside the chamber:

$$\frac{A_c}{A_t} = 0.1 * \left[\frac{1+\frac{\gamma-1}{2}*0.1^2}{\frac{\gamma+1}{2}}\right]^{\frac{\gamma+1}{2(\gamma-1)}} \quad (6)$$

The chamber area can be double-checked with the following "rule of thumb" formula [3]:

$$\varepsilon_c = \frac{A_c}{A_t} \simeq (8 * D_t^{\ -0.6} + 1.25) \quad (7)$$

The chamber length can be found by dividing the characteristic length by A_c/A_t ratio. This length can be double checked with the equation below. A characteristic length of ~1.2m is assumed and needs to be confirmed with further testing.

$$L_c \simeq \frac{1}{\varepsilon_c}\left[L^* - 1/3 * \sqrt{\frac{A_t}{\pi}} * \frac{\varepsilon_c * \sqrt{\varepsilon_c} - 1}{tan(\theta_c)}\right] \quad (8)$$

A conical nozzle was chosen for simplicity. The nozzle contraction half-angle is 30 degrees and the nozzle half-angle is 15 degrees, complying with accepted standards [2].

The parameters for the cooling system are calculated sequentially. First the engine is split into a series of circular cross-sections. Next, the diameter of the inner wall D is calculated using the above equations and simple trigonometry. Following which the temperature T and pressure P of each section are found via Matlab's built-in isentropic flow tables [5]. Before finding the heat transfer rate needed for the cooling system, the viscosity of the combustion gases must be calculated based on the temperature of each section. This can be done using the Sutherland equation [6].

$$\frac{\mu}{\mu_0} = \left(\frac{T}{T_0}\right)^{3/2} * \frac{T_0+S}{T+S} \tag{9}$$

Each constituent gas in the exhaust has its own unique μ_0, T_0 and S value which must be used in this equation. The viscosities of each constituent gas can then be added up based on their mass fractions to get the final viscosity. Now the heat flux from the combustion gases can be calculated using the Bartz equation [7].

$$h_g = \frac{0.026}{D_t^{0.2}}\left(\frac{P_c}{c^*}\right)^{0.8}\left(\frac{D_t}{D}\right)^{1.8} C_p \mu^{0.2}\left(\frac{T}{\frac{T+Tw}{2}}\right)^{0.8-0.2w} \tag{10}$$

$$q = h_g(T_{aw} - T_w) \tag{11}$$

In these formulas, h_g is the convective heat transfer coefficient from the hot gases into the chamber wall. T_{aw} is the adiabatic wall temperature of the gases, which is approximately 0.94x the normal exhaust gas temperature. T_w is the maximum allowable temperature of the chamber wall, which is dependent on the material used and the required yield stress. Copper is used as the chamber wall for this engine due to its high thermal conductivity, and the fact that it is relatively cheap and easy to machine. Therefore a maximum temperature of 750K is used, which results in a yield strength σ_y of approximately 80 Mpa [8]. The minimum wall thickness t for the following section is then calculated with the hoop stress formula. A factor of safety of 2 is applied for this engine. The temperature of the outer chamber wall (the wall which touches the coolant) is then determined.

$$t = \frac{P*(D/2+t)}{2*\sigma_y}*\mathrm{FOS} \tag{12}$$

$$T_{w,\,inner} = T_w - q * t/k \tag{13}$$

The size of the cooling jacket around the engine then needs to be determined. For maximum efficiency this cooling jacket diameter ($D_{coolant}$) varies in each section, similar to the wall thickness. A simple coaxial shell is used for this engine's cooling jacket. The convective heat transfer inside each section of the cooling jacket is determined with the following formulae.

$$Re = \rho_{coolant} * v_{coolant} * D_h/\mu_{coolant} \tag{14}$$

$$Nu = Re^{0.8} * Pr^{1/3} * 0.023 \tag{15}$$

$$h_c = Nu * k_{coolant}/D_h \tag{16}$$

$$q_{coolant} = h_c * (T_{w,\,inner} - T_{coolant}) * 2\pi(D + 2t) \tag{17}$$

In order for the engine to stay at a stable temperature, $q_{coolant}$ must be the same as q from the bartz equation. Therefore the diameter of the cooling jacket must be iterated until the two heat transfer values match.

$$D_{coolant} = D_{coolant} * q_{coolant}/q \tag{18}$$

The pressure drop in this section can then be calculated using the Haaland equation.

$$f = \left(\left(\frac{-1}{1.8}\right)log\left[\left(\frac{\frac{\epsilon}{D_h}}{3.7}\right)^{1.11} + \frac{6.9}{Re}\right]\right)^2 \tag{19}$$

$$\Delta P_{coolant} = \frac{f*\rho_{coolant}*v_{coolant}^2*L}{2*D_h} \tag{20}$$

$q_{coolant}$ must then be added to the coolant to increase its temperature. The above formulae should be run for each section of the engine. For this engine, calculations were done from the bottom up (ie- coolant enters at the bottom of the nozzle and exits near the injector).

A splash-plate injector is planned for use in this design due to its simplicity, ease of manufacture and good atomization. Formulae for the design of this injector are taken from the work of Inamura and Yanaoka [9], and from Sarchami, Ashgriz and Tran [10]. Injector design will begin once these formulas have been successfully replicated.

2.2 Propellant System

The design goal behind the propellant feed system was safe and effective delivery of propellant to the combustion chamber. To achieve this, mechanical failsafes have been implemented to maintain control of the system in the event of electronics failure. The propellant system is mounted to a 80/20 aluminum frame. The engine thrust is to be beared and measured by a welded steel structure.

The propellant feed system is pressure-fed and uses high-pressure nitrogen gas as a pressurant for the ethanol and nitrous oxide run tanks. Components are connected using high-pressure stainless steel tubing. Estimations of the run tank pressures have been performed through computations of the theoretical pressure loss between each run tank and the injector. The predicted ethanol and nitrous oxide run tank pressures are 7.79 MPa and 7.90 MPa respectively. Valves have been implemented to manage pressures, control propellant flow, and permit the system to vent when required. The flow coefficient, fail-safe capabilities, actuation method, pressure rating, seal compatibility and cost were considered when selecting each valve. The main valves of the propellant feed systems required for a normally closed configuration with minimal pressure drop. Pneumatically actuated ball valves were selected as the MFV (main fuel valve) and MOV (main oxidizer valve) due to their high flow coefficients and spring-return capabilities. The pneumatic systems for both valves operate at 689 kPa

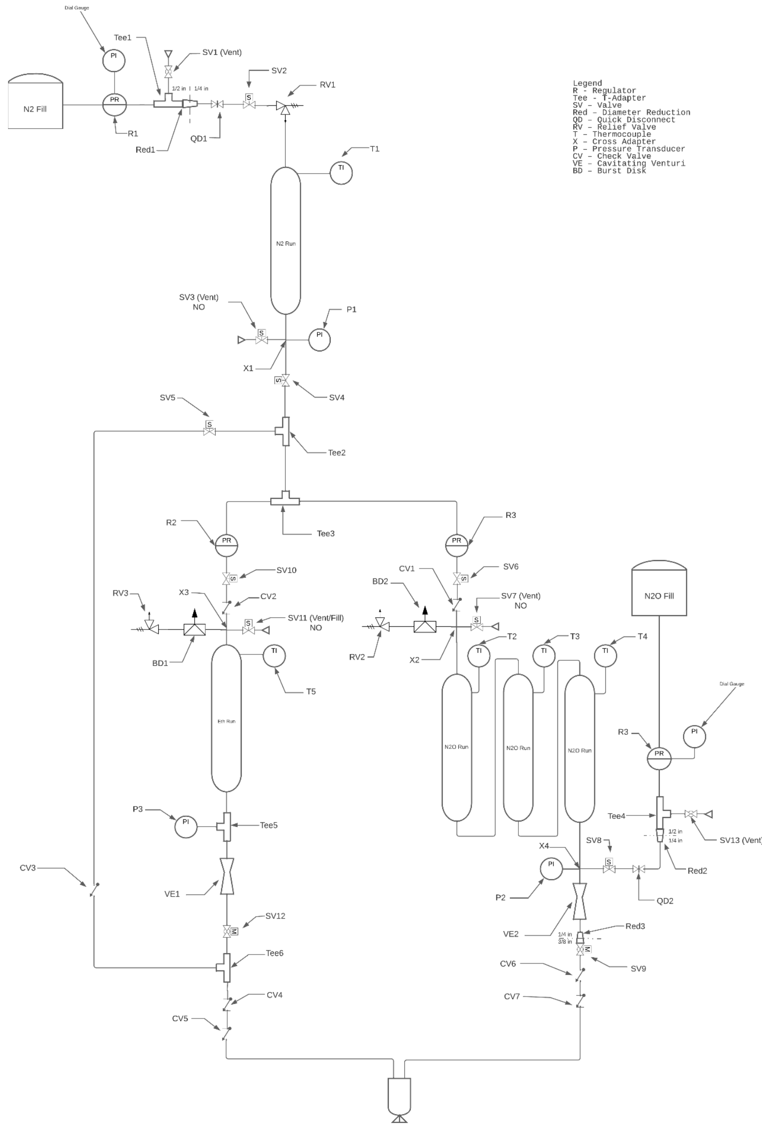

Figure 1: Propellant System P&ID

Each control and isolation valve in the pressure feed system must have a normally closed configuration with a high pressure rating. Electrically actuated solenoid valves were selected on this basis. The vent valves in the pressure feed system were positioned to provide each reservoir with a direct venting pathway. Normally open electric solenoid valves were therefore selected for the vent valves. Relief valves and burst disks have been implemented to prevent over-pressurization and manage regulator creep. Additional check valves with a 6.89 kPa cracking pressure and poppet configuration have been implemented in the propellant, pressurant and purge lines to prevent backflow.

The flow control mechanism for each feed line is to be a cavitating venturi. Cavitating venturis were selected due to their ability to act as a passive flow controller. Ethanol is to be poured from an elevated reservoir into its run tank in a depressurized state. Nitrogen pressurant and nitrous oxide are to be loaded into the run tanks through a detachable line of regulators, solenoid and vent valves. The nitrous oxide fill tank is to be inverted to minimize the amount of vapor that is transferred to the run tank. System verification is to be accomplished through cold flow testing with liquid carbon dioxide and water to simulate the flow of nitrous oxide and ethanol respectively.

2.3 *Avionics and Ground Station Systems*

The avionics and ground station are a pair of systems that work together to control the timing of the combustion reaction in the Borealis engine and abort from any dangerous engine states. The avionics is directly attached to the engine, and controls the propellant system valves and combustion ignition sequence using its own control loop and abort conditions. It also measures the rocket engine state and sends that data wirelessly to the ground station. The ground station is located a safe distance away from the static fire test area for personnel to monitor the rocket engine, and is disconnected from all rocket engine subsystems other than the electrical power supply. It displays any data that the avionics broadcasts, and sends commands to modify the control loop of the avionics and allows for ground station personnel to send abort commands to the avionics when an unsafe state is detected. In the event that both the abort systems fail, a physical disconnect switch will be used to shut off power to the igniter and all propellant valves to force them into their fail-safe states.

The avionics system is designed to operate its control loop at a minimum of 100 times per second, which includes the following: estimating the state of the rocket from its sensors, controlling the actuators in the propellant system and the igniter, reporting the estimated rocket state back to the ground station system, and listening for commands from the ground station. The control loop runs on an 8 bit 16MHz microcontroller (Atmel ATmega2560). The microcontroller is directly connected to two analog to digital converters (Texas Instruments ADS1115) for sensor measurements, several solid state relays to control the propellant valves, and a radio transceiver (2.4GHz DIGI XBee) to communicate with the ground station. The analog to digital converters measure the thermocouples (T1 to T5) and pressure transducers (P1 to P3) connected to the propellant system in the P&ID diagram. T1 and T5 measure the temperature for N2 and Ethanol. T2, T3, T4 measure the temperature of N2O. P1, P2, P3 measure the pressure of the N2, N2O, and Ethanol tanks. P2 and P3 also measure the flow rate at the cavitating venturis. The ground station system uses the same microcontroller and radio transceiver as the avionics system.

3. Results and Discussion

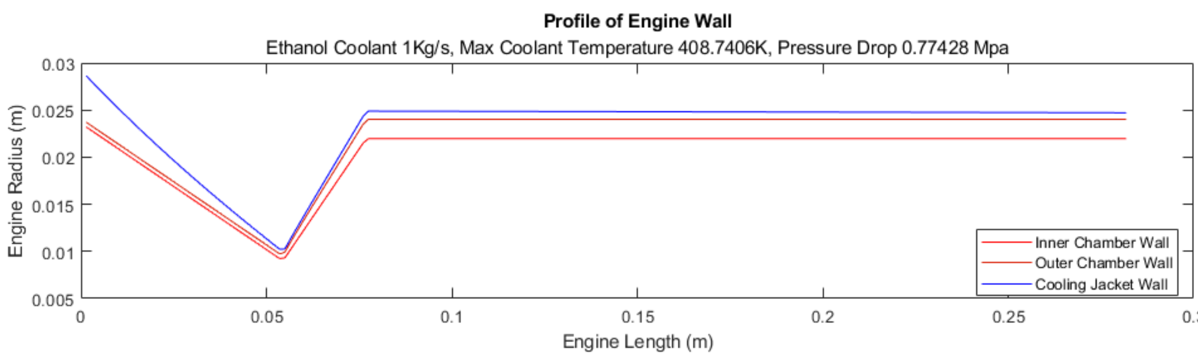

Figure 2: Engine Wall Graph for Ethanol

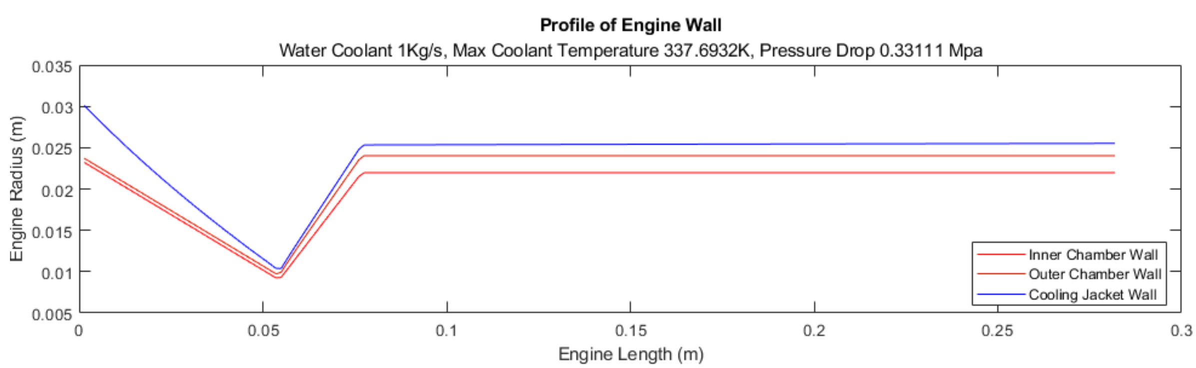

Figure 3: Engine Wall Graph for Water

The engine produced from the above formulae and values has a nozzle radius of 23.6mm, a throat radius of 9mm and a combustion chamber radius of 22mm. The combustion chamber is relatively long at 205mm. The wall thickness in the chamber is 2.1mm, but thins out quickly near the throat and nozzle as pressure decreases. The wall thickness was set to a minimum of 0.5mm to ensure machinability and prevent the engine from being too brittle. The heat flux at the throat reaches approximately $2.5*10^7$ W/m^2. This necessitates a cooling jacket with a thickness of only 0.66mm at the throat, but this thickness increases to 1.5mm near the chamber.

Both ethanol and water were tested as potential coolants for this engine. At the engine's 0.19Kg/s flow rate, the ethanol will reach temperatures higher than the wall temperature before reaching the end of the engine, meaning that pure regenerative cooling is unfeasible for an engine of this size and an external coolant flow is needed. Water is significantly better than Ethanol at cooling the engine due to its high thermal conductivity, specific heat capacity and higher density [6][12][13]. This calculation confirms that water undergoes a significantly lower temperature increase and pressure drop.

4. Concluding Remarks

The research behind RPG's Borealis Engine has been detailed throughout this report. A rocket engine was successfully parametrically designed, and the propellant feed and avionics systems were successfully sourced. This project is under progress as a way to participate in the Launch Canada 2022 competition for a static-fire test, while simultaneously using it to ensure a successful transference of knowledge to the next batch of RPG members. Future plans for the engine subsystems involve the use of linear solenoid valves that can offer finer propellant flow control than an on/off state, a closed loop control system in the avionics, and the use of a modular redundant avionics system to reduce the impact of a single electrical subsystem failure.

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