# NovaTech Space Systems: NOVA 1

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**Abstract**

This document details the design process of the NOVA 1 rocket. This rocket was designed to deliver a 1000 kg payload to 500km LEO with a $\Delta$V budget of 9 km/s. This rocket ultimately costs $13.5 million per launch, has a launch mass of 15,266 kg, and produces a net $\Delta$V of 9.175 km/s. The final burnout altitude is 545 km with a complete burn time of 439.6 s. Further research is to be conducted on the self landing capabilities of the 1st stage, however its superior performance allows plenty of fuel budget for such maneuvers.

## 1 Nomenclature

$P_a$ = Atmospheric Pressure

$P_i$ = Injector Pressure

$P_c$ = Chamber Pressure

$P_{fe}$ = Fuel Exit Pressure

$P_{oxe}$ = Oxidizer Exit Pressure

$P_{t,e}$ = Turbine Exit Pressure

$P_{c,gg}$ = Gas Generator Chamber Pressure

$T_c$ = Chamber Temperature

$T_e$ = Exit Temperature

$T_{co}$ = Initial Film Coolant Temperature

$T_{cgg}$ = Gas Generator Chamber Temperature

$T_{wg}$ = Maximum Gas Wall Temperature

$\gamma_c$ = Chamber Heat Capacity Ratio

$\gamma_{gg}$ = Gas Generator Heat Capacity Ratio

$A_c$ = Chamber Area

$A_t$ = Throat Area

$A_e$ = Exit Area

$L^*$ = Combustion Chamber Length

$\mathfrak{M}$ = Exhaust Molar Mass

$V_e$ = Exit Velocity

$C_{eq}$ = Equivalent Velocity

$C^*$ = Characteristic Velocity

$C_F$ = Coefficient of Thrust

$F$ = Thrust

$C_{p,gg}$ = Molar Heat Capacity Divided by Universal Gas Constant

$m_0$ = Total Mass

$m_p$ = Propellant Mass

$m_e$ = Structural Mass

$\dot{m}$ = Mass Flow Rate

$\dot{m}_{ox}$ = Oxidizer Flow Rate

$\dot{m}_f$ = Fuel Flow Rate

$\dot{m}_{gg}$ = Gas Generator Mass Flow Rate

$G_c$ = Film-Coolant Weight Flow Per Unit Area Of Cooled Camber Wall

$h_g$ = Gas-Side Heat-Transfer Coefficient

$I_{sp}$ = Specific Impulse

$\beta$ = Payload Factor

$\epsilon_1$ = Structural Factor

$\epsilon_2$ = Expansion Ratio

$\eta_c$ = Film-Cooling Efficiency

$\eta_t$ Turbine Efficiency

$\eta_{pf}$ Fuel Pump Efficiency

$\eta_{pox}$ Oxidizer Pump Efficiency

$\phi$ = Mixing ratio

$r$ = Mixture ratio

$r_{stoich}$ = Stoichiometric Mixture ratio

$r_{gg}$ = Gas Generator Mixture ratio

$T$ = Torque

$C$ = Cost

$C_{pvc}$ = Average Specific Heat at Constant Pressure

$CR$ = Contraction Ratio

$R_u$ = Universal Gas Constant

$W$ = Weight

$\alpha$ = Mass Cost Constant

$\omega$ = Rotation Speed

# 2 Introduction

The NOVA 1 rocket was prepared for a rocket design challenge put forward by a set of anonymous venture capitalist. The rocket was designed to deliver a 1000 kg payload to an altitude of 500 km. The rocket was additionally specifically required to be low-cost, and to have a reusable first stage. The venture capitalists additionally provided a set of assumptions that the rocket should follow:

- The cost of each launch can be calculated from the following equation:

$$C = \alpha * m_0[\frac{1-\beta_1}{N_1} + \sum_{j=2}^{\infty}(1-\beta_j)] \tag{1}$$

- For stages that use liquid bi-propellant engines assume a structural factor of $\epsilon = 0.15$
- Assume the total ΔV of the mission is 9.0 km/sec

To meet these requirements the NOVA 1 design process primarily utilized optimization scripts to determine optimum mass and ΔV distributions, fuel selection, and choice design components such as $\phi$, $\epsilon_2$, and $P_c$. Additional research was performed to finalize the design of cyrogentic storage system, turbomachinery, and cooling systems. The temporary design for the recovery system consists of deploying parachutes to assist the 1st stage in landing safely, how ever future work is proposed to develop propulsive reentry/landing of the 1st stage. This paper will summarize the design process of this rocket, with an in depth analysis of the cryogentic fuel storage, turbomachinery, combustion chamber, and nozzle design with concluding remarks and suggested future work.

# 3 Technical Approach

## 3.1 Staging/Motor Selection

The initial design of the rocket begin with determining the optimal number of stages to deliver a payload to LEO. Taking inspiration from existing launch vehicles, such as Space X's Falcon 9 rocket, and based on common practice in the space industry a 2-stage design was chosen [5]. Next the selection of a Solid Rocket Motor (SRM) vs a Liquid Rocket Motor (LRM) was undertaken. While solid rocket motors can produce vast amounts of thrust, liquid rocket engines typically produce high $I_{sp}$ values, leading to a more efficient rocket, and allow for precise throttling of the rocket motors. The team decided to select liquid rocket motors for both stages for the aforementioned reasons. Specifically, staged combustion LRMs were selected due to their high launch performance.

## 3.2 Fuel Selection

A trade study for fuel and oxidizer selection was then undertaken with high attention given to the \$ to $I_{sp}$ ratio of various propellant combinations. The different oxidizer and fuels used are taken from existing combinations and are validated using CEA analysis [2].

The combinations of oxidizers and fuels are shown in Tab.1 highlighting the specific impulse that is produced using CEA.

| Oxidizer | Fuel | Specific Impulse [sec] |
|---|---|---|
| Liquid Oxygen | Liquid Hydrogen | 145.06 |
| | Kerosene | 117.74 |
| | Liquid Methane | 118.7 |
| Liquid Fluorine | Liquid Hydrogen | 167.7 |
| | Liquid Methane | 130.25 |
| | Hydrozine | 152.29 |
| Nitrogen Textroxide | Liquid Hydrogen | 56.46 |
| | Liquid Methane | 111.64 |
| | Kerosene | 106.77 |
| | Hydrozine | 115.5 |
| Nitric Acid | Liquid Hydrogen | 123.85 |
| | Liquid Methane | 106.25 |
| | kerosene | 102.95 |

Table 1: CEA analysis of oxidizer and fuel combinations

The results showed that using $LO_2$ combined with $H_2$, as well as $LF_2$ combined with $LH_2$ gave the highest $I_{sp}$ values while also being readily available. The next factor that was considered was cost using current market value of both the oxidizers and fuels considered. Research showed that the ratio of $I_{sp}$ to dollar was $56.14\frac{sec}{\$}$ for $LO_2 + LH_2$ and $5.63\frac{sec}{\$}$ for $LF_2 + LH_2$ [8]. Through this study liquid Hydrogen and liquid Oxygen were selected for their affordability, availability, and low environmental impact.

## 3.3 Driving Performance Parameters Selection

Once the propellant selection had been finalized, the chamber pressure $P_c$, mixing ratio $\phi$, and expansion ratio $\epsilon_2$ were determined through iterating various value combinations to maximize the $I_{sp}$ of the rocket using NASA's CEA software. The finalized values can be see in Tab.2:

| Stage | $P_c$ [MPa] | $\phi$ | $\epsilon_2$ |
|---|---|---|---|
| 1 | 30.4 | 2.1 | 25 |
| 2 | 30.4 | 2.1 | 50 |

Table 2: CEA iteratively determined design values.

A $\phi$ of 25 was chosen for the first stage to compromise between performance losses due to over expansion of the nozzle at sea level, and $I_{sp}$ losses. The first stage burns out at an altitude of 155 km where the ambient pressure is  0 Pa, resulting in no need to pressure match so $\epsilon_2$ was unrestricted for the second stage and could be maximized to achieve a large $I_{sp}$. The altitude estimation used in determining the $\epsilon_2$ values for stage 1 was made using (Andrew talk about altitude estimation script, include the equations used and a table with final burn times and altitude burnout for each stage)

### 3.4 Additional Fuel Performance Parameters

With $P_c$, $\phi$, $\epsilon_2$ finalized the following fuel performance parameters could be given by CEA as seen in Tab.4. Further calculations and design choices are guided by these values.

| **Variables** | **1st Stage** | **2nd Stage** |
|---|---|---|
| $P_c$ | 30.4 [MPa] | 30.4 [MPa] |
| $P_e$ | 0.0895 [Mpa] | 0.0340 [MPa] |
| $T_c$ | 2878 [K] | 2878 [K] |
| $T_e$ | 855.7 [K] | 673.6 [K] |
| $\mathfrak{M}$ | 9.61 [amu] | 9.61[amu] |
| $C^*$ | 2417 [m/s] | 2417 [m/s] |
| $M_e$ | 4.21 | 4.88 |
| $C_F$ | 1.72 | 1.79 |
| $\gamma_c$ | 1.22 | 1.22 |

Table 3: CEA calculated fuel performance parameters.

### 3.5 Optimum Mass Calculations

To get an idea of what the mass of each stage will be, we utilized a similar problem found in an orbital mechanics textbook. Using the problem as a guideline, a MatLab script was produced to show us the mass distribution between the two stages. Inputting our known $I_{sp}$ for each stage as well as the $\epsilon$ values provided into the script, we then compared it to our desired $\delta$V. Leveraging our launch location of French Guiana, we expect a $\delta$V contribution for the surface of $0.46\frac{km}{sec}$. From here we knew we only desired a $\delta$V of $8.54\frac{km}{sec}$ between the two stages. Using these known values, the MatLab script showed that the optimum mass of the first and second stage are $4.1345\frac{km}{sec}$ and $4.399\frac{km}{sec}$ respectively.

## 4 Cryogenic Fuel Storage

The storage temp for our liquid Hydrogen and liquid Oxygen is 20°K and 90°K respectively. To minimize boil off during launch and throughout the mission, the NOVA 1 utilizes a double walled vacuum storage system for the cryogenic propellants. The inner wall of the storage system will utilize SA240 Grade 304 stainless steel for structural purposes as stainless steel is strong and naturally corrosion resistance. The between wall vacuum will be filled with evacuated glass bubbles held under a mild vacuum to reduce the transfer of heat via radiation between the walls. The outer wall will be constructed out of SA516 Grade 70 carbon steel, and will act primarily as a insulator layer. This design took heavy inspiration from the large-scale hydrogen storage tanks employed at the NASA Kennedy Space Center [6]. Additional design considerations to reduce boil off include painting the exterior of the rocket body white to reduce absorption from solar radiation due to our launch location being in close proximity to the equator, and the 2nd stage employing a polyethylene radiation shield to further minimize solar radiation outside of atmosphere.

The orientation of the tanks in both stages will involve the Oxygen tank resting in tandem on top of the Hydrogen tank to lower the rockets center of gravity as the storage tanks drain. All storage tanks will be cylindrical in shape and have an inner radius of 2 m. In the first stage will hold 6469.56 kg of Hydrogen and 3080.742 kg of Oxygen, with the second stage holding 1744.829 kg of Hydrogen and 830.871 kg of Oxygen. The total estimated height of the propellant tanks is 9.5 m.

# 5 Turbomachinery

## 5.1 Turbopump Sizing

To find the mass of the turbopump the pressure of fuel exiting the pump and the pressure of oxidizer exiting the pump were estimated using the two equations below. $P_c$ was equal to 300 atm so $P_{fe}$ was equal to 546 atm and $P_{oxe}$ was equal to 336 atm.

$$P_{fe} = (1 + 0.82)P_c \tag{2}$$

$$P_{oxe} = (1 + 0.12)P_c \tag{3}$$

Using the minimum pressure from the two above equations the pressure of the gas generator combustion chamber was estimated below, this equation yielded an estimated pressure of 329.4 atm.

$$P_{cgg} = \frac{P_{oxe}}{1.02} \tag{4}$$

To continue the estimation the turbine exit pressure was estimated in the next equation, this equation yielded an estimated pressure of 1.84 atm.

$$P_{t,e} = P_a * (\frac{\gamma_{gg}+1}{2})^{\frac{\gamma_{gg}}{\gamma_{gg}-1}} \tag{5}$$

The next step taken to find the mass of the turbopump was to estimate the power required for its turbine using the equation below. The value of $\dot{m}_{gg}$ was unknown, but $T_{cgg}$ was set to 1500 K and $C_{p,gg}$ was found to be 7.873 $\frac{kJ}{kg*K}$ using CEARUN. $\gamma_{gg}$ was determined to be 1.317, also using CEARUN. All efficiencies to follow are estimated to be 50%.

$$TurbinePower = \dot{m}_{gg}\eta_t C_{p,gg}T_{cgg}[1-(\frac{P_{t,e}}{P_{cgg}})^{\frac{\gamma_{gg}-1}{\gamma_{gg}}}] \tag{6}$$

To solve for the unknown $\dot{m}_{gg}$ value, the following equation was used with the above equation. All known values were filled in so $\dot{m}_{gg}$ could be solved for. The value for $r_{gg}$ was found to be 1.185 using CEARUN. $\Delta P_f = P_{fe}$ and $\Delta P_{ox} = P_{oxe}$ were estimated to be such values by taking the inlet pressure to be insignificant. $\rho_f = 70.9$ $\frac{kg}{m^3}$ and $\rho_o x = 1141$ $\frac{kg}{m^3}$ were found to be the densities of liquid hydrogen and liquid oxygen respectively. For stage 1 of the rocket $\dot{m}_{ox1} = 37.157$ $\frac{kg}{s}$ and $\dot{m}_{f1} = 9.7526$ $\frac{kg}{s}$. For stage 2 of the rocket $\dot{m}_{ox2} = 9.2596$ $\frac{kg}{s}$ and $\dot{m}_{f2}$ $= 2.4304\frac{kg}{s}$.

$$TurbinePower = (\dot{m}_f + \frac{\dot{m}_{gg}}{1+r_{gg}}) * \frac{\Delta P_f}{\rho_f * \eta_{pf}} + (\dot{m}_{ox} + \frac{r_{gg} * \dot{m}_{gg}}{1+r_{gg}}) * \frac{\Delta P_{ox}}{\rho_{ox} * \eta_{pox}} \tag{7}$$

The result of finding $\dot{m}_{gg}$ gave a value of $\dot{m}_{gg1} = 5.035$ $\frac{kg}{s}$ for stage 1 and $\dot{m}_{gg2} = 1.255$ $\frac{kg}{s}$ for stage 2. The power required for the turbopump of stage 1 was 21198 kW and the power for the second stage turbopump was 5283.7 kW. Assuming a rotation speed of $\omega = 50000$ $\frac{rad}{s}$ the equation below was used to determine the torque.

$$\tau = \frac{TurbinePower}{\omega} \tag{8}$$

The torque of the two stages was found to be $\tau_1 = 423.96$ N*m or 312.69 ft*lbs and $\tau_2 =$ 105.67 N*m or 77.940 ft*lbs. Finally, using imperial units for torque the below equation yields the estimated weight of the turbopump in pounds for both stages.

$$W = 5.26 * \tau^{0.638} \tag{9}$$

This equation yielded a weights of $W_1 = 205.5$ lbs or 93.21 kg for stage 1 and $W_2 = 84.71$ lbs or 38.42 kg for stage 2.

## 5.2 Injector Pressure

The injector pressure for liquid rocket engines tends to range from 5% to 50% of the chamber pressure. For this rocket design the estimated injector pressure was 8.36 MPa, or 27.5% of chamber pressure.

## 5.3 Turbomachinery Overview

The turbopumps for stages 1 and 2 are expected to have masses of 93.21 kg and 38.42 kg respectively. Looking as the overall mass of turbomachinery for similar rockets, namely the Vinci rocket, the estimated total turbo machinery mass, including tubing and valves, was about 160 kg for stage 1 and about 70 kg for stage 2. The expected length of the rocket this machinery would occupy roughly ranges from 1.25 to 1.5 meters for the first stage and 0.75 to 1.25 meters for the second stage. The combustion chamber pressure of the two stages is the same so the expected injector pressure is 8.36 MPa for both of them.

# 6 Combustion Chamber: Film Cooling

To manage the high combustion chamber temperatures (2878 K) and prevent melting the stainless steel combustion chamber (1783K). Gaseous film cooling is implemented in the design. Hydrogen supplied by the turbine is used for this task along with the outer parameter of injectors being fuel only. The starting temperature of this fluid is 922K and enough flow rate of hydrogen will keep it to a max of 1355k up to the throat section. 1355K was picked for margin of safety and integrity of the steel. The theoretical equation from Hatch and Papell is used to find out the film coolant weight flow rate per unit area.

$$(T_c - T_{wg})/(T_c - T_{co}) = e^{-(h_g/(G_c * C_{cpv}))*\eta_c} \tag{10}$$

This is then used to multiply it to the area of the combustion chamber to get .118 kg/s of extra fuel during flight to maintain ideal temperatures of the wall temperature. Due to both stages being similar this applies to both stages

# 7 Nozzle

## 7.1 Nozzle Type

To get the best performance at each stage of the mission, a bell nozzle is selected for each of the two mission stages. The design was chosen in order to reduce the losses due to divergence. Additionally using method of characteristics, a bell nozzle is able to cancel out negative effects of expansion and compression shocks. Bell nozzles are known for being long, heavy, and hard to manufacture, however these effects are negated by our excess propulsive power and being able to reuse the first stage. Having a reusable first stage allows us to have a steeper initial manufacturing cost, but reduce all subsequent costs.

## 7.2 Nozzle Sizing

To find area of of the nozzle throat, the relation between the burn time and the mass flow rate was first used. The equation below shows how the burn time, mass flow rate, and fuel performance parameters are related.

$$t_b = \frac{m_p}{\dot{m}} = \frac{m_p * I_{sp} * g_o}{F} \tag{11}$$

Once the burn time was found, the known propellant mass was used to find the mass flow rate through the nozzle. This results in a value of $\dot{m} = 44.48\frac{kg}{sec}$ for the first stage and $11.724\frac{kg}{sec}$ for the second stage. For each stage, the mass flow rate was plugged into the following equation to find the area of the throat.

$$t_b = P_c A_t \sqrt{\frac{\gamma \mathfrak{M}}{R_u T_c}} (\frac{2}{\gamma + 1})^{\dfrac{\gamma+1}{2(\gamma-1)}} \tag{12}$$

Using the known fuel performance values, the throat area for the first and second stages are found to be $0.00373m^2$ and $0.000929m^2$ respectively. Next, the known expansion ratio of each stage is used along with the throat area to determine the area of the exit. Using the relation the exit area is found to be $0.0933m^2$ and $0.04645m^2$ for the first and second stages respecively.

## 7.3 Chamber Sizing

The use of common propellants allowed us to lean on tabulated combustion chamber values, such as table 8.1 in the textbook. This table showed that for our selected propellants of $LO_2$ and $LH_2$, a range for L* could range from 0.7m-1m. A combustion chamber length of 1m was chosen as a larger combustion chamber has an effect on making the system more energetically efficient. Once L* was found, the contraction ratio, CR, was determined using existing rockets for reference. For rockets that have a single engine thrust of 150,000 lbf, a CR of 1.6 was used. Scaling this to our thrust of 187,000N and 49,000N for the first and second stages respectively, we get a CR of 5.7 for the first stage and 8 for the second stage. The equation for the CR was then used as shown below was used to find the chamber area of each stage.

$$CR = \frac{A_c}{A_t} \tag{13}$$

Using the above equation, a chamber area of $A_{c1} = 0.0212m^2$ and $A_{c2} = 0.00743m^2$ was found. The material being used for the chamber is stainless steel which has a melting point of 1,783k. Due to our chamber temperature being over 1,000k above the melting point, cooling techniques must be applied to the interior of the chamber to keep its structural integrity.

## 7.4 Cooling Selection

We chose a single-pass regenerative cooling system for our rocket. We will be taking into account initial rocket parameters such as initial pressure, mass, chamber pressure, exit velocity, and temperature are specified. As well as, parameters for the regenerative cooling system, including the equivalent diameter, channel diameter, wall thickness, and fuel mass flow rate, are defined.

## 7.5 Coolant Jacket

The number of channels ($N$) and the mass flow rate through each channel ($\dot{m}_{\text{chan}}$) are determined. Resulting in values of approximately 18.1719 and 2.4486 kg/s,

$$\dot{m}_{\text{chan}} = \frac{1}{N} m_{\text{f}} \quad \text{and} \quad N = \pi^{1/2} D_e + 0.8(d + 2t_w) \tag{14}$$

## 7.6 Heat Transfer

Taking into account the heat transfer within the cooling system is an important part of maintaining optimal performance and preventing thermal damage. The equation below shows the exchange of thermal energy between the gas and the wall surface.

$$q' = h_g \cdot (T_r - T_{Wg}) \tag{15}$$

The gas-side wall temperature ($T_{Wg}$) stabilizes at 2602.4952 K after 9 iterations. With this value, the convective heat transfer coefficient ($h_g$) is calculated iteratively using the Newton-Raphson method and Nusselt number. The liquid-side wall temperature ($T_{Wl}$) converges after 2 iterations to a final value of 2501.7347 K.

$$Nu = 0.023 \times (Pr^{0.3}) \times (M^{0.33}) \quad \text{and} \quad h_g = Nu\frac{k_{\text{gas}}}{D_e} \tag{16}$$

Now, the convective heat transfer coefficient for the liquid coolant ($h_l$) and the new liquid temperature and pressure for the cooling system. Subsequently, the new liquid temperature ($T_l$) and pressure ($P_l$) are computed as 2300.0739 K and 296.2 Pa.

These final numbers from the regenerative cooling system analysis are important for ensuring the success and safety of our rocket. By determining the new liquid temperature and pressure, we first need the convective heat transfer coefficient for the liquid coolant ($h_l$), this will help maintain the nozzle's temperature within safe limits. Optimizing the number of cooling channels ($N$) as well as the mass flow rate ($\dot{m}_{\text{chan}}$) will assist in preventing overheating and damage to any components by enhancing heat transfer. Therefore, by determining the liquid temperature ($T_l$) and pressure ($P_l$), we can accurately assess heat absorption and ensure safe operation. These values will then influence the efficiency of the overall propulsion system, impacting gas expansion within the engine.

## 7.7 Additional Nozzle Oerformance Parameters

| **Variables** | **1st Stage** | **2nd Stage** |
|---|---|---|
| $P_e$ | 0.089 [MPa] | 0.034 [MPa] |
| $T_e$ | 856 [K] | 674 [K] |
| $\epsilon_2$ | 25 | 50 |
| $A_t$ | 0.00373 [K] | 0.000929 [K] |
| $A_e$ | 0.0933 [amu] | 0.0465[amu] |
| $V_e$ | 2306 [m/s] | 2374.5 [m/s] |
| $C_{eq}$ | 4164 | 4315 |
| $C_F$ | 1.73 | 1.79 |
| $F$ | 0.187 [MN] | 0.0494 [MN] |
| $\dot{m}$ | 44.5 [kg/s] | 11.72 [kg/s] |
| $I_{sp}$ | 429 [s] | 440 [s] |

Table 4: CEA calculated nozzle performance parameters.

# 8 Rocket Overview / Future Work

The NOVA 1 is estimated to have a base radius of roughly 2 m, and a height of 16.5 m, with a correctional area of 12.566 $m^2$. To determine the actual $\Delta$V delivered by the rocket the following equation is used:

$$\Delta V = g_0 I_{sp}(\ln[\epsilon(1-\beta_1)+\beta_1] + \ln[\epsilon(1-\beta_2)+\beta_2]) \tag{17}$$

Using this equation the actual $\Delta$V of the rocket is found to be 9.175 km/s. Accounting for our surface velocity $V_{surf}$ at launch, this results in the rocket producing an excess $\Delta$V of 0.632 km/s. The first stage of the rocket will burnout after 214.7 s, with a burnout altitude of 155 km. The second stage will burnout after an additional 224.9 s, at an altitude of 545 km. Using the cost estimation equation provided by the venture capitalists, a final cost per launch of the rocket can be found to be $13.5 million, using an $\alpha$ of $1000/kg.

If the entirety of the fuel is consumed in the launch process then the rocket will greatly overshoot its desired final altitude. It is instead proposed that a portion of the fuel be reserved for the propulsive landing of the 1st stage. As previously mentioned in the introduction, the temporary design for the controlled reentry of the first stage is via parachutes that will be deployed after the first stage has detached. This method has several flaws including the reentry trajectory being uncontrollable. A propulsive landing system similar to that seen in the Falcon 9 would be optimal for this rocket, and given the excessive $Delta$V this rocket is able to produce, there is more then enough room in the design for such a system. Additional future work proposed for the rocket include performing an in depth structural analysis of the cryogenic storage tanks and the combustion chamber to determine their respective thicknesses.

# 9 Conclusion

The NOVA 1 rocket is a 2-stage high performance rocket design to deliver a 1000 kg payload to 500 km LEO. The rocket costs $13.5 million per launch, and produces 9.125 km/s of $\Delta$V. The rocket uses liquid Hydrogen and Oxygen, which are stored using a double walled vacuum storage method, and are fed into its staged combustion LRMs in each stage through use of a turbo-pump. The chamber and nozzle are cooled using a combination of film and regenerative cooling methods. Future research is to be performed on the thickness of the storage tanks, and combustion chamber. Finally, if the rocket receives funding a propulsive landing system will be researched for the 1st stage using the rokcets excess fuel.

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