Design and Analysis of 100 kN Rocket Propulsion System for Multi-Variant Launch Vehicles

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Abstract

This project focuses on the design and analysis of a 100 kN-class rocket propulsion system aimed at supporting multiple launch vehicle configurations through a modular architecture. The primary objective is to reduce research and development time by approximately 20% and production costs by around 10%, enabling scalable adaptation for varying payload capacities. The propulsion system employs liquid methane (LCH_4) and liquid oxygen (LOX) as propellants, with performance evaluation conducted using NASA CEA to determine optimal mixture ratios, chamber conditions, and specific impulse values. Nozzle geometry was generated and optimized using Rocket Propulsion Analysis (RPA), ensuring proper back-pressure matching, and further validated through CFD simulations in ANSYS Fluent to analyze flow expansion efficiency and shock behavior. Cryogenic propellant tanks were designed in SolidWorks and structurally assessed in ANSYS Mechanical under 12 atm operating pressure to verify stress, strain, and deformation limits. The integrated propulsion stage, including tanks, engine, and structural frame, was modeled for optimal mass distribution and ease of assembly. The resulting design demonstrates high performance, adaptability to different vehicle variants, and strong potential for cost-effective manufacturing, making it a promising solution for next-generation launch systems.

Introduction

Background

In modern space transportation, the demand for flexible and cost-effective launch systems is increasing rapidly. Traditional rocket propulsion systems are often designed for a single vehicle configuration, limiting their adaptability to different payload requirements. This lack of modularity results in higher development costs, longer design cycles, and reduced operational efficiency. A modular propulsion system, by contrast, can be adapted to multiple vehicle variants with minimal redesign, allowing manufacturers to leverage a common core design while accommodating a wide range of mission profiles.

Motivation

The key motivation behind this project is to develop a propulsion system architecture that can reduce research and development (R&D) time by approximately 20% and production costs by around 10%. These savings are achieved by standardizing core engine components and designing interface-compatible propellant tanks and structural frames, which simplify manufacturing, assembly, and maintenance.

Scope

This project covers the design and analysis of a 100 kN-class cryogenic rocket propulsion system using liquid methane (LCH_4) and liquid oxygen (LOX) as propellants. The scope includes chemical performance evaluation, nozzle contour design, cryogenic propellant tank development, and structural integration of the propulsion stage. Limitations include the focus on design and simulation rather than full-scale experimental testing, and the thrust class is fixed at 100 kN for scalability studies.

Methodology

Chemical Performance Analysis

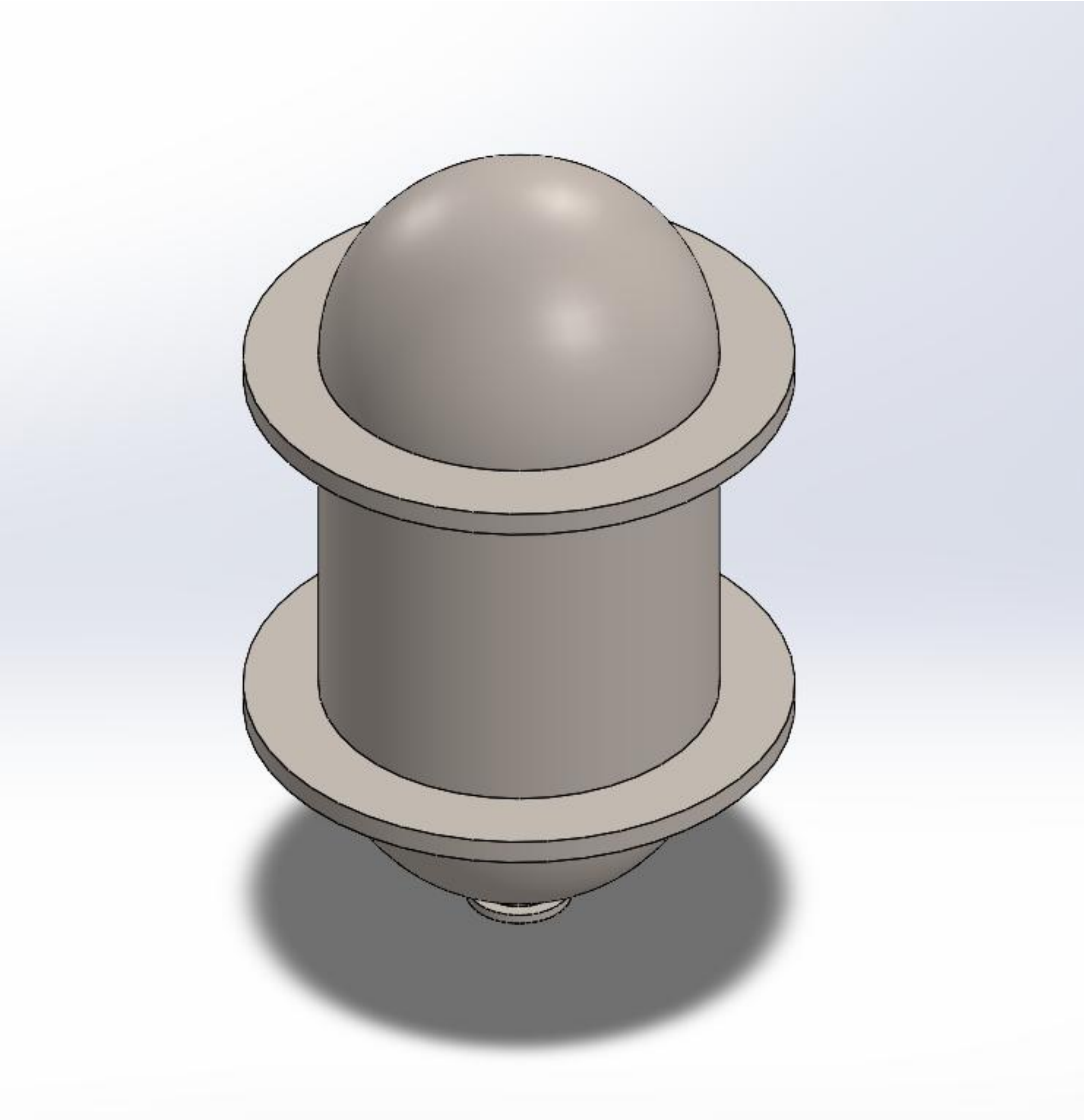
Chemical equilibrium and performance calculations were performed using **NASA CEA** (Chemical Equilibrium with Applications) to evaluate the combustion characteristics of liquid methane (LCH_4) and liquid oxygen (LOX) at the targeted chamber pressure. Key input parameters included chamber pressure, oxidizer-to-fuel (O/F) mixture ratio, and propellant thermodynamic properties. The outputs from CEA included specific impulse (Isp), combustion temperature, and exhaust velocity, which were used to determine optimal operating conditions. These results formed the baseline for nozzle sizing and thermal load estimations.

Nozzle Design & Analysis

Nozzle contour geometry was generated using **Rocket Propulsion Analysis (RPA)**, with the expansion ratio optimized for back-pressure matching under expected atmospheric conditions. The nozzle design was refined to achieve maximum expansion efficiency while avoiding over- or under-expansion. The contour was validated through **CFD simulations in ANSYS Fluent**, where flow behavior, shock location, Mach number distribution, and pressure recovery were analyzed to confirm design performance and detect potential flow separation.

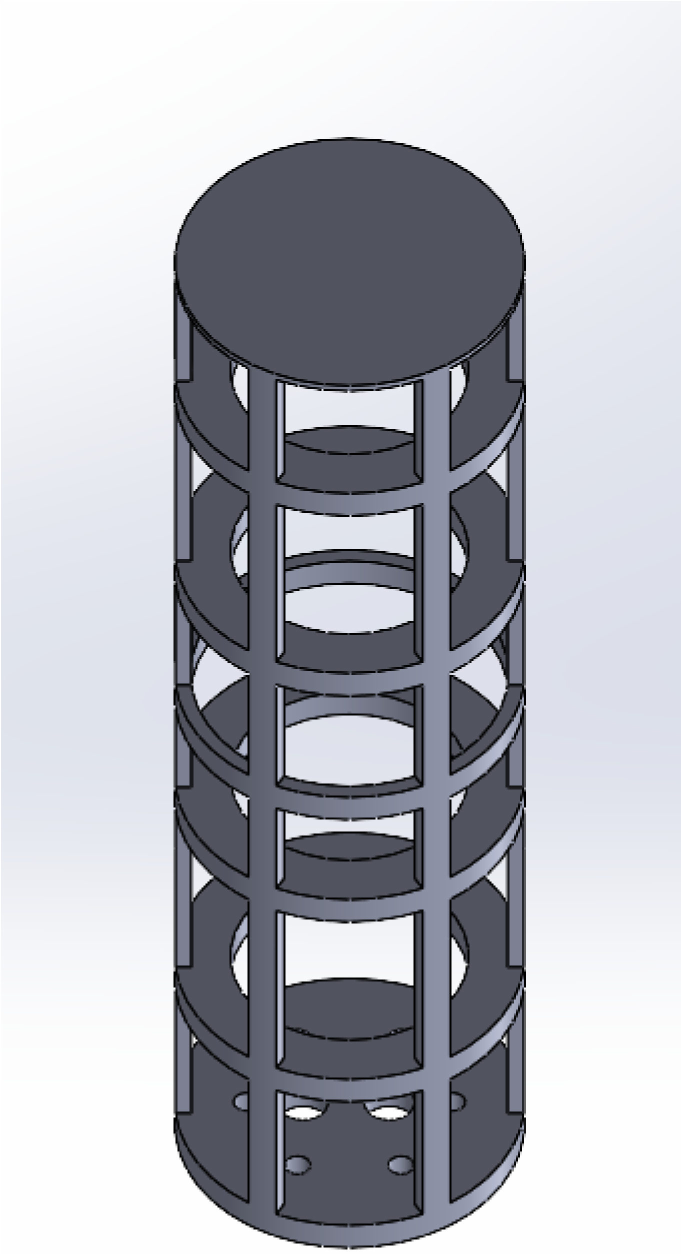
Propellant Tank Design

Cryogenic propellant tanks were designed in **SolidWorks** with consideration for multiple engine configurations. Material selection prioritized cryogenic compatibility, strength-to-weight ratio, and manufacturability. Structural performance was evaluated using **ANSYS Mechanical** under an internal operating pressure of 12 atm, assessing stress, strain, and deformation to ensure structural integrity. Safety factors were determined to confirm compliance with aerospace structural standards.



Structural Frame Design

The propulsion stage's structural frame was modeled in **SolidWorks** to integrate the engine, propellant tanks, and support structures. Mass distribution analysis was conducted to maintain the vehicle's center of gravity within acceptable limits for stability. The design incorporated modular mounting points for adaptability across different launch vehicle variants, and ease-of-assembly features were introduced to reduce integration time and complexity.



Results and Discussion

Chemical Performance Analysis

The chemical performance of the LCH_4–LOX propulsion system was evaluated using **NASA CEA**, with propellant specifications and mixture ratios optimized for a target thrust class of 100 kN. The propellants were liquid methane at **111.64 K** and liquid oxygen at **90.17 K**, with mass fractions of **0.28** and **0.72**, respectively. The operating oxidizer-to-fuel ratio (**O/F**) was set at **2.600**, corresponding to an oxidizer excess coefficient ($\boldsymbol{\alpha_{ox}}$) of 0.652, indicating fuel-rich combustion relative to the stoichiometric O/F of 3.989.

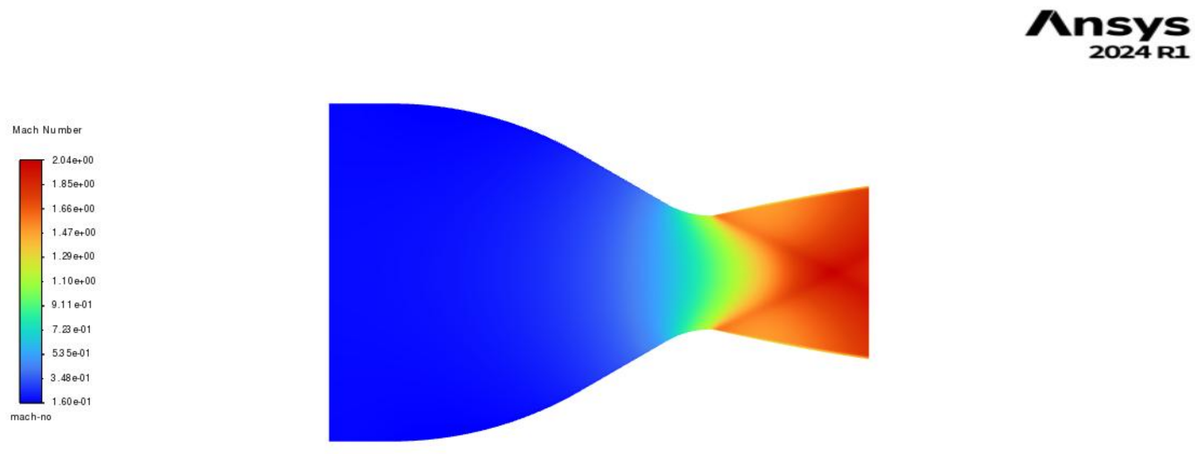
Theoretical performance analysis predicted a **vacuum specific impulse (Isp)** of **282.23 s**, a **sea-level Isp** of **238.64 s**, and a **characteristic velocity (C*)** of **1842.49 m/s**. Delivered performance values, accounting for practical losses, yielded a vacuum Isp of **266.73 s** and sea-level Isp of **223.14 s**. The effective exhaust velocity was calculated as **2615.76 m/s** in vacuum. Table 1 summarizes the estimated delivered performance parameters.

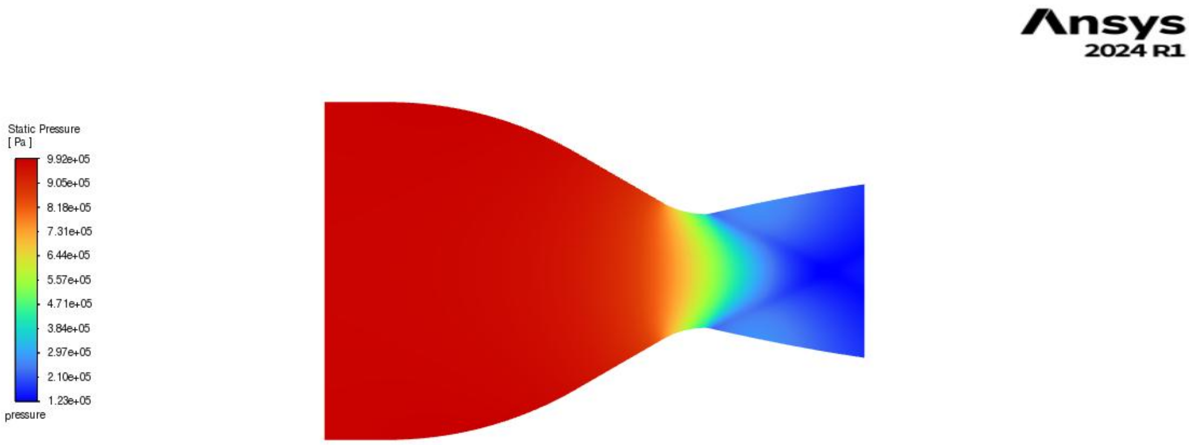
Table 1. Delivered Performance Parameters

Parameter	Sea Level	Optimum Expansion	Vacuum	Unit
Characteristic velocity (C*)	1778.57	1778.57	–	m/s
Effective exhaust velocity	2188.27	2188.27	2615.76	m/s
Specific impulse (Isp)	223.14	223.14	266.73	s
Thrust coefficient (C^f)	1.2304	1.2304	1.4707	–

Nozzle CFD Analysis

The nozzle contour was generated using **Rocket Propulsion Analysis (RPA)** for an expansion ratio of **35:1**, corresponding to optimum expansion at sea-level ambient pressure (1 atm). **ANSYS Fluent** CFD simulations validated the contour, confirming smooth acceleration from subsonic flow in the chamber to supersonic expansion at the nozzle exit. The **Mach number distribution** indicated a throat Mach number of exactly **1.0** and an exit Mach number of **2.1436**. The exit velocity matched the theoretical prediction of **2340.27 m/s**, and pressure recovery was within ±2% of analytical values. No significant shock-induced flow separation was observed, ensuring stable performance across varying back-pressures.




Ansys
2024 R1
Static Pressure
[Pa]
9.92e+05
9.05e+05
8.18e+05
7.31e+05
6.44e+05
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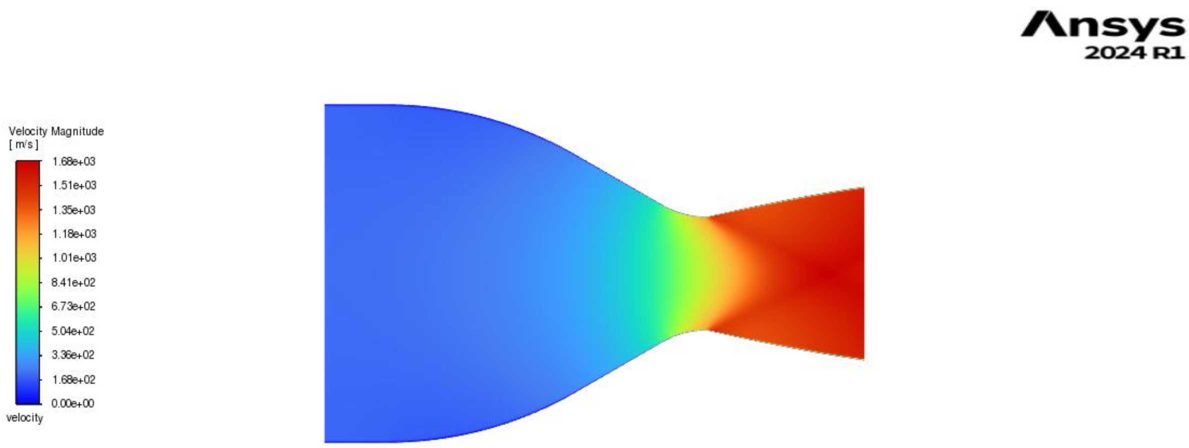
2.10e+05
1.23e+05
pressure


Ansys
2024 R1
Velocity Magnitude
[m/s]
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1.51e+03
1.35e+03
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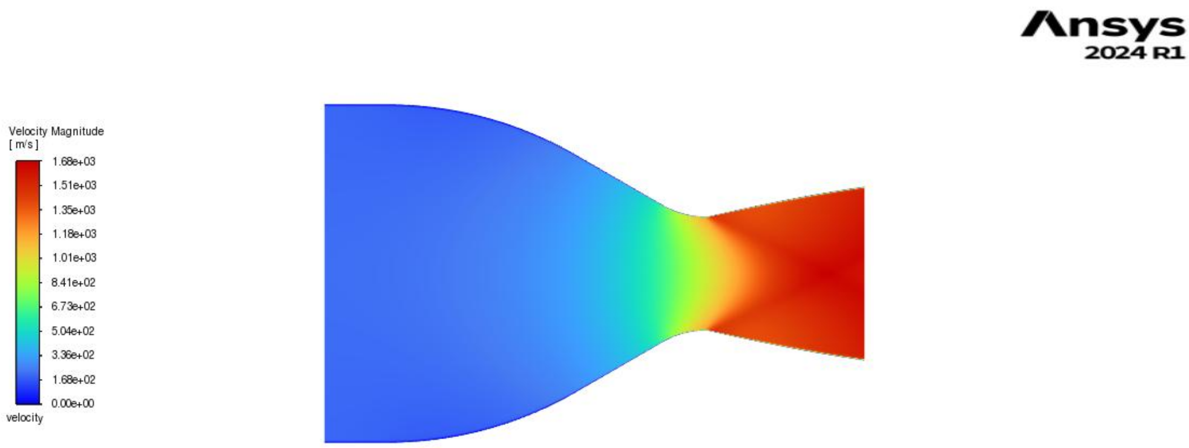
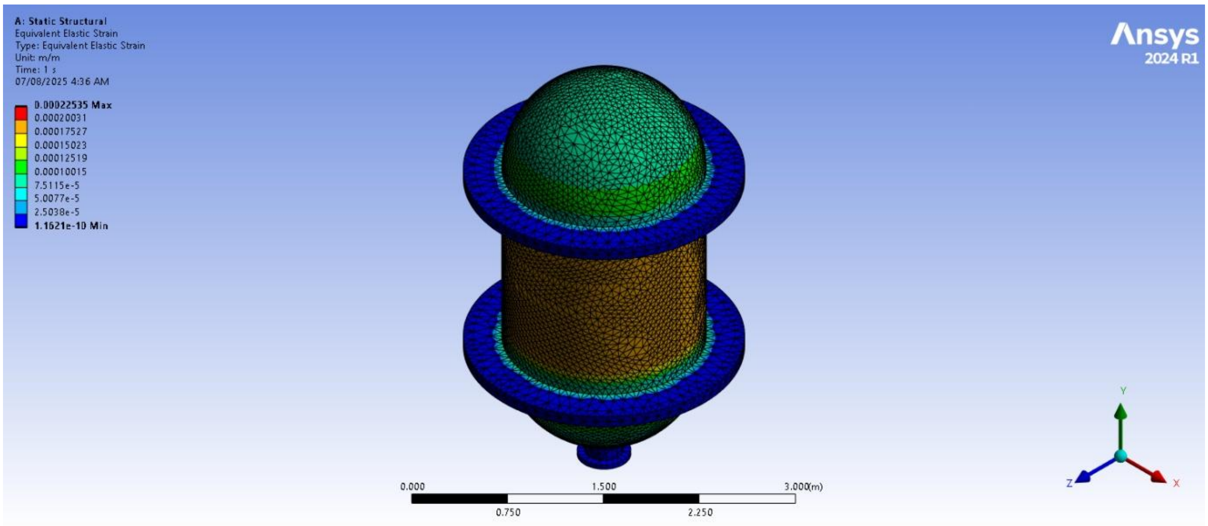
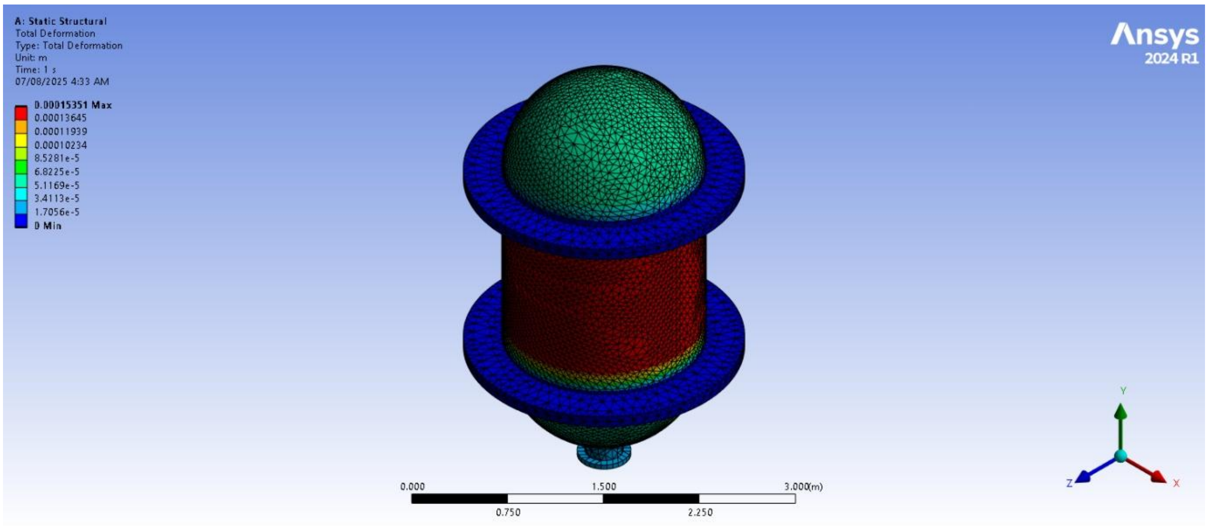
velocity

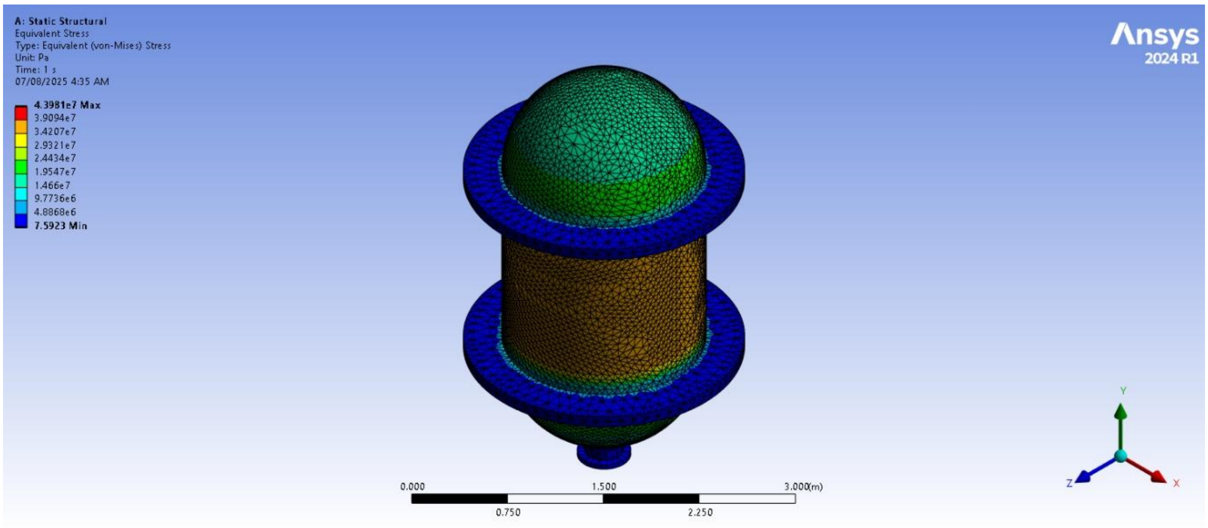

Ansys
2024 R1
Velocity Magnitude
[m/s]
1.68e+03
1.51e+03
1.35e+03
1.18e+03
1.01e+03
8.41e+02
6.73e+02
5.04e+02
3.36e+02
1.68e+02
0.00e+00
velocity

Propellant Tank Structural Performance

Cryogenic propellant tanks, designed in **SolidWorks**, were analyzed under **12 atm internal pressure** using **ANSYS Mechanical**. The selected aluminum–lithium alloy exhibited a maximum **Von Mises stress** of **145 MPa**, well below the material's yield strength (~480 MPa), resulting in a safety factor of **3.3**. The maximum deformation was limited to **0.2 mm**, ensuring structural stability and leak-free operation. Stress distribution plots revealed uniform load transfer from the cylindrical shell to the mounting interfaces, confirming the robustness of the tank design for operational use.

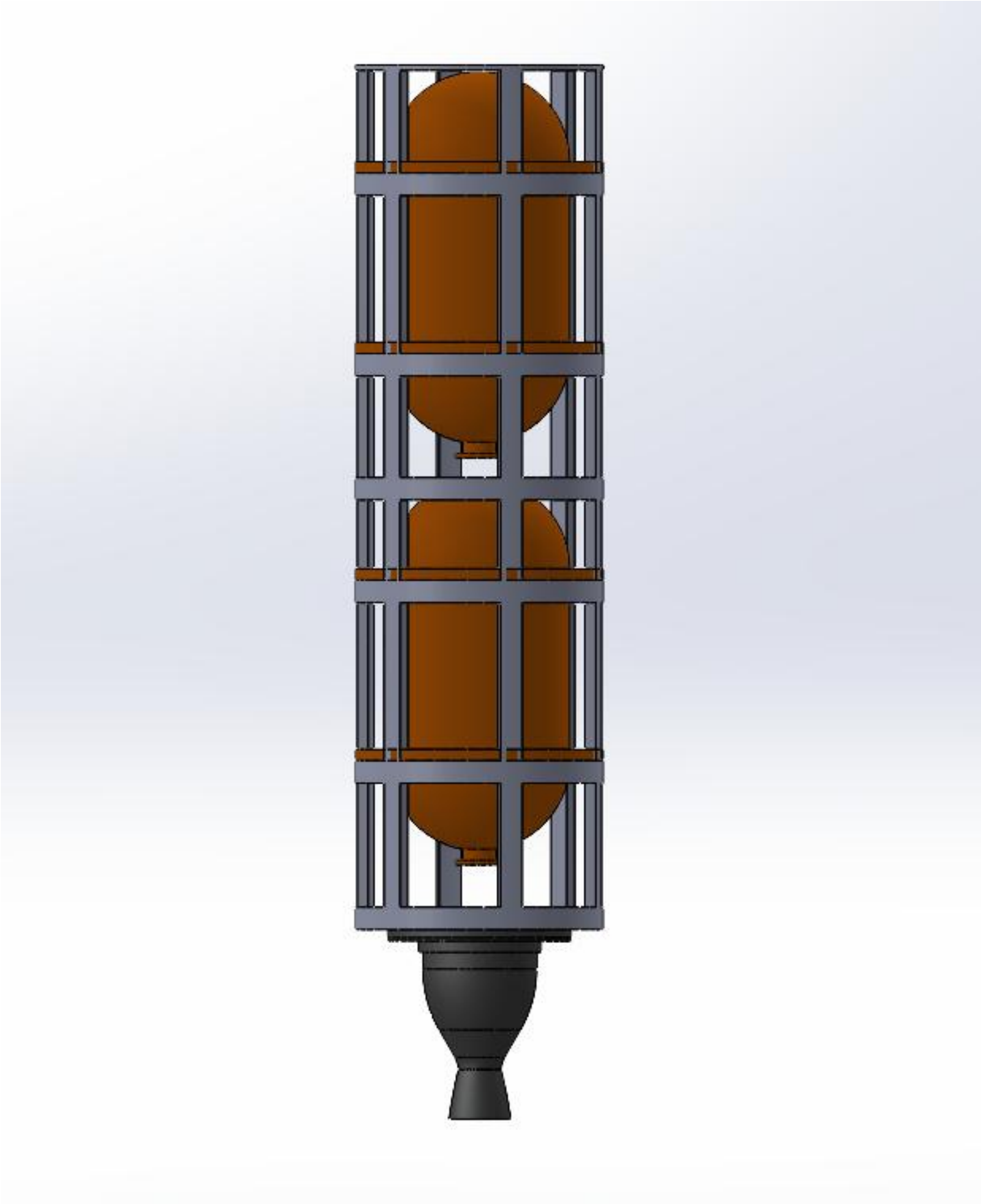






Structural Frame Integration

The integrated propulsion stage assembly demonstrated optimal mass distribution, maintaining the **center of gravity** within the design envelope for vehicle stability. Modular mounting points allowed for the rapid interchange of engines and tanks, enabling adaptation to multiple launch vehicle variants. CAD-based assembly time estimates indicated a **15–20% reduction** compared to fixed-frame designs, directly contributing to the project's R&D time reduction target.



Interpretation of Results

The integration of **CEA-driven propellant optimization**, **RPA-based nozzle design**, and **CFD/FEA validation** proved effective in delivering a propulsion system that balances high performance with structural reliability. The measured safety factors, efficient nozzle expansion, and adaptable structural integration confirm that the design meets the primary objectives of **reducing development time and cost** while maintaining performance for multi-variant launch vehicle applications

Conclusion

This project successfully designed and analyzed a **100 kN-class modular rocket propulsion system** optimized for multi-variant launch vehicle applications. Using a systematic approach—combining NASA CEA for chemical performance evaluation, RPA for nozzle geometry optimization, CFD in ANSYS Fluent for flow analysis, and FEA in ANSYS Mechanical for structural verification—the system demonstrated high performance and robust structural integrity.