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Case Report

Evaluation of Propellant Slosh Effects and Baffle Requirements in a Hybrid Rocket Oxidizer Tank

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Abstract

This paper investigates the necessity of incorporating internal baffles in the oxidizer tank of a vertically launched hybrid rocket employing a regressive fuel grain configuration. The study begins with a review of potential sources of rocket instability, with particular emphasis on propellant slosh and its influence on vehicle stability throughout the flight mission. A systems engineering methodology is adopted to define the mission objectives and establish the corresponding design requirements. These requirements are subsequently decomposed into lower-level requirements and verified through analytical methods to ensure compliance with the overall system objectives. The propulsion system performance is evaluated using NASA Chemical Equilibrium with Applications (CEA) software, considering Hydroxyl-Terminated Polybutadiene (HTPB) as the fuel and Nitrous Oxide (N_2O) as the oxidizer at an oxidizer-to-fuel ratio of 8 and a combustion chamber pressure of 30 bar. To characterize the operational environment, the NRLMSIS atmospheric model is employed to obtain atmospheric temperature, air density, and gravity data for 5 May 2026 over an altitude range from sea level to 20 km with 1 km intervals. Based on the derived requirements, preliminary oxidizer tank design activities are conducted, including tank sizing, material selection, wall-thickness determination, and the definition of internal tank features. The dynamic effects of oxidizer motion during flight are analytically evaluated and compared with the stabilizing forces generated by tank ullage pressure. The results demonstrate that the forces induced by oxidizer slosh are significantly smaller than the available stabilizing forces and therefore have a negligible impact on vehicle stability. Consequently, the study concludes that internal baffles are not required for the proposed oxidizer tank design, enabling a simpler and lighter tank configuration without compromising flight stability.

Keywords

Oxidizer Slosh, Baffles, Regressive Fuel Grain, Deceleration

1. Introduction

Rocket stability is the inherent or controlled capability of a rocket vehicle to maintain or return to its desired attitude and trajectory following a disturbance, through the action of restoring aerodynamic, propulsive, or control-system moments. [1]

Loss of stability in a rocket can occur due to a variety of interacting factors associated with propulsion, aerodynamics, structural dynamics, and vehicle mass properties. One of the primary causes is combustion instability, which arises when pressure oscillations within the combustion chamber grow rapidly and become self-sustaining. This can result from poor mixing of fuel and oxidizer, injector design deficiencies that

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create uneven propellant distribution, coupling between chamber acoustic modes and combustion processes, or delayed combustion reactions that generate feedback oscillations. These phenomena can lead to significant fluctuations in chamber pressure and thrust, potentially causing performance degradation or structural damage. [14]

Another major source of instability is flight (aerodynamic) instability, which occurs when the aerodynamic forces acting on the vehicle can no longer maintain stable flight. This often happens when the center of pressure (CP) moves ahead of the center of mass (CM), creating a destabilizing aerodynamic moment. Additional contributors include excessive wind shear, atmospheric disturbances, inadequate fin effectiveness in smaller rockets, failures in guidance, navigation, or control systems, thrust vector control malfunctions, and uneven engine thrust or engine-out conditions. Such factors can cause excessive attitude deviations and loss of trajectory control.

Propellant slosh instability is also a significant concern, particularly in vehicles with large, partially filled propellant tanks. The movement of liquid propellant inside the tanks can generate dynamic forces and moments that interact with the vehicle's attitude-control system. If not adequately controlled through tank baffles or damping devices, the sloshing motion can couple with vehicle dynamics and induce oscillatory behavior that degrades flight stability. [13]

A further source of instability is structural or dynamic instability, commonly referred to as Pogo oscillation. This phenomenon results from the interaction between pressure fluctuations in the propellant feed system and the structural vibrations of the launch vehicle. When engine thrust oscillations resonate with the rocket's natural structural frequencies, large longitudinal vibrations can develop, potentially threatening vehicle integrity and mission success.

Feed-system instability can also contribute to overall rocket instability. Problems such as cavitation in turbopumps, vapor formation within propellant feed lines, pressure regulator malfunctions, and dynamic coupling between injectors and the feed system can cause fluctuations in propellant flow rates and chamber pressure. These disturbances may propagate throughout the propulsion system and lead to unstable engine operation.

Finally, changes in mass properties during flight can affect vehicle stability. As propellant is consumed, the center of gravity shifts and the vehicle's moments of inertia change continuously. If these changes are not properly accounted for in the control system design, the effectiveness of the guidance and control laws may be reduced, leading to degraded stability margins and increased difficulty in maintaining the desired flight path. Together, these factors highlight the complex and multidisciplinary nature of rocket stability and the importance of careful design, analysis, and testing throughout the development process. [15]

In this section only vehicle instability cause by Propellant Slosh is studied with defined design parameters.

Designing the internal features of a rocket propellant tank goes far beyond simply storing fuel; it involves carefully managing fluid behavior under extreme conditions such as high acceleration, microgravity, and significant temperature gradients. An effective design must strike a balance between structural integrity, mass efficiency, and reliable propellant delivery. One of the primary internal features includes baffles, also known as slosh control structures, which are internal plates or rings used to reduce fluid movement during acceleration, stage separation, or attitude changes. By limiting sloshing, they help maintain vehicle stability, although increasing their number can improve performance at the cost of added weight. [2]

Another critical component is the propellant management device (PMD), especially important in microgravity environments where traditional gravity-fed systems are ineffective. PMDs ensure that propellant remains near the outlet using mechanisms such as vanes that guide liquid via surface tension, sponges or screens that trap liquid, and channels that direct flow, all of which rely on capillary action. The ullage space, or the empty volume above the liquid propellant, also plays a key role and is maintained by a pressurization system that ensures consistent propellant flow to the engines. This system typically uses inert gases like helium and helps prevent cavitation in turbopumps. Anti-vortex devices are installed near the tank outlet to prevent the formation of vortices that could draw gas into the feed line, thereby ensuring steady engine operation. Thermal management features are equally important, particularly for cryogenic propellants such as liquid oxygen or hydrogen, and may include internal insulation, control of temperature stratification, and sometimes active mixing systems to maintain uniform conditions. In addition, sensors and instrumentation are embedded within the tank to monitor parameters such as propellant level, temperature, and pressure, which are crucial for both control and safety. Finally, the geometry of the outlet and feed system is carefully designed to maximize propellant usage, often incorporating sump regions to collect remaining fuel and minimize unusable residuals. [2]

Inside a rocket tank, liquid propellant doesn't stay still. During acceleration, or engine cutoff, it sloshes. This can: Shift the center of mass, Couple with vehicle dynamics (called sloshing instability) and Destabilize guidance and control systems. Baffles reduce this motion by dissipating energy and breaking up fluid flow. Among a few standard configurations: Ring baffles: Circular plates with holes, mounted inside the tank, Radial baffles: Plates extending inward from the tank wall, Perforated plates: Flat plates with carefully sized holes and Vaned baffles: Angled structures that redirect flow. Typical situations where rockets may avoid baffles: Tanks are nearly full all the time, very small launch vehicles, Spin-stabilized upper stages and Use of diaphragms, bladders, PMDs (propellant management devices), or surface-tension systems instead. [16] Typical situations where baffles become mandatory: Large cryogenic oxidizer tanks, long slender rockets, Vehicles with strong pitch/yaw maneuvers, Partial-fill conditions and restart able upper stages or microgravity operations. Baffles are usually placed: Near the tank mid-height or at nodes of expected slosh modes (determined via simulation). This ties into modal analysis of the fluid system. Structural loads: Baffles must survive: Launch acceleration (several g's), Dynamic pressure from moving fluid and Thermal stresses (especially in cryogenic tanks). Adding baffles improves stability but introduces costs: Increased mass (bad for payload capacity), Manufacturing complexity and Potential for trapping gas or creating uneven draining. [3]

2. Design Requirement

Table 1. Expression Of Interest with Higher Level Requirement.

Customer Expression of Interest	Level 1 Requirement
We are seeking an experienced oxidizer tank designer/manufacturer for a hybrid rocket propulsion system using Nitrous Oxide (N_2O) as the oxidizer and HTPB as the fuel. The oxidizer tank shall have a capacity of 4.8 liters with a filling pressure of 51 bar and must operate reliably within a vehicle altitude range of 0–20 km. The propulsion system requires a 1-inch diameter propellant feed line with an oxidizer flow rate of 0.3 kg/sec. The tank design should withstand the vehicle operating velocity and associated aerodynamic and structural loads during vertical takeoff and flight. The design should also ensure structural integrity, ease of manufacturing, lightweight construction, and compatibility with a powered flight duration of approximately 20 seconds.	R-1: Types Of Propellant(oxidizer), Nitrous Oxide (N_2O). R-2: Types Of Propellant(fuel), HTPB. R-3: Amount Of Oxidizer in Litter, 4.8 Liter. R-4: oxidizer Filled Pressure in Tank, 51 Bar. R-5: Vehicle Operation Altitude Domain, (0-20) Km. R-6: Propellant Feeding Pipe Dimension 1 Inch Diameter And 0.3 Kg/Sec Propellant Flow Rate. R-7: Vertical Takeoff. R-8: Powered Vehicle Flight Time (20 Sec) R-9: The Tank Can Withstand Vehicle Velocity.

Table 2. Derived requirements.

No.	Level 1 Requirement	Level 2 Requirement	Level 3 Requirement	Verification
1	R-1: Types of Propellant(oxidizer), Nitrous Oxide (N_2O).	R-11: vehicle cycle should exclude unnecessary component from the system. R-12: tank material must be compatible with nitrous oxide.		
2	R-2: Types Of Propellant(fuel), HTPB.			
3	R-3: Amount Of Oxidizer in Litter, 4.8 Liter.	R-31: oxidizer tank volume should consider Nitrous Oxide properties and filling temperature and pressure. R-32: practical diameter to length ratio of tank should be consider. R-33 tank volume should consider ullage space		
4	R-4: oxidizer Filled Pressure in Tank, 51 Bar.	R-41: The Ambient Temperature Should Not Exceed 20 ℃ While the Oxidizer Is Stored in The Tank. R-42: Tank thickness must be calculated with enough safety factor.		
5	R-5: Vehicle Operation Altitude Domain, (0-20) Km.	R-51 System should consider atmosphere temperature up to 20 km. R-52 System consider air density up to 20 km R-53 System consider gravitational force up to 20 km		
6	R-6: propellant feeding pipe dimension 1 inch diameter and 0.3 kg/sec propellant flow rate.			

No.	Level 1 Requirement	Level 2 Requirement	Level 3 Requirement	Verification
7	R-7: Vertical Takeoff.			
8	R-8: powered vehicle flight time (20 sec)			
9	R-9: vehicle deceleration deceleration up to 500 m/sec.			

R-1: Types Of Propellant, Nitrous Oxide (N_2O)

R-11 Vehicle Cycle Should Exclude Unnecessary Component from The System.

No need to include thermal management device since Nitrous Oxide is not cryogenic propellants.

No need to add inert gases like helium, since Nitrous Oxide by itself pressurization.

N_2O in compressed-gas cylinders is present in both the liquid and gaseous states. N_2O cylinders are factory filled to 90% to 95% capacity with liquid N_2O. Above the liquid in the tank is N_2O gas. The gas pressure within the cylinder of N_2O is approximately 52 bar at 25 ℃. The pressure of the N_2O vapor floating above the liquid N_2O is 52 bar. As the gaseous N_2O exits from the cylinder, liquid N_2O vaporizes to replace it. The pressure of this "new" gas is 52 bar. This process continues, liquid N_2O converting to gaseous N_2O, with the gas pressure remaining at 52 bar, until no more liquid remains to replace the gas. [2]

R-12 Tank Material Must Be Compatible with Nitrous Oxide.

6061 aluminum sheets (6061-T651)

Most commonly recommended materials are aluminum.

alloys or stainless steel, depending on the design priorities (weight, cost, manufacturability, and pressure requirements). [4]

R-2: Types Of Propellant(fuel), HTPB

As seen from below figure HTPB with N_2O at 0/f 8 generate chamber pressure 30 bar.

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            THEORETICAL ROCKET PERFORMANCE ASSUMING EQUILIBRIUM

         COMPOSITION DURING EXPANSION FROM INFINITE AREA COMBUSTOR

Pin =    435.1 PSIA
CASE = GEDLU

           REACTANT                         WT FRACTION      ENERGY      TEMP
                                             (SEE,NOTE)     KJ/KG-MOL      K
OXIDANT    N2O                               1.0000000     81401.672    293.000
FUEL       HTPB                              1.0000000    -51900.000    293.000

O/F=    8.00000 %FUEL= 11.111111  R,EQ.RATIO= 1.111278  PHI,EQ.RATIO= 1.111591

                 CHAMBER   THROAT     EXIT     EXIT     EXIT     EXIT     EXIT     EXIT
Pinf/P            1.0000   1.7362   24.000   26.000   28.000   30.000   32.000   20.611
P, BAR            30.000   17.279   1.2500   1.1538   1.0714   1.0000  0.93750   1.4555
T, K             3366.27  3179.67  2389.77  2364.41  2340.56  2318.66  2296.62  2437.00
RHO, KG/CU M    2.9029.0 1.7924.0 1.8076-1 1.8076-1 1.5640-1 1.4840-1 1.4194-1 2.0604-1
H, KJ/KG         1661.83 -1051.14 -1124.40 -1124.83 -1821.28 -1861.25 -1003.24 -1003.01
U, KJ/KG          563.39   87.126 -1800.94 -1888.29 -1861.29 -1840.59 -1967.05 -1709.44
G, KJ/KG        -21134.3 -29970.3 -24339.0 -24516.9 -23976.5 -23633.7 -23639.9 -24702.5
S, KJ/(NG)(K)     9.7249   9.7249   9.7249   9.7249   9.7249   9.7249   9.7249   9.7249
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Figure 1. *Results for Chemical Combustion from NASA CEA, Software.*

NASA CEA (Chemical Equilibrium with Applications) is a computer program developed by the National Aeronautics and Space Administration for analyzing the chemical and thermodynamic performance of rocket propellants, combustion processes, and high-temperature reacting gases. [10]

R-3: Amount Of Propellant in Litter, 4.8 Liter

R-31: oxidizer tank volume should consider Nitrous Oxide properties and filling temperature and pressure.

Size Of Tank

$$\mathrm{volume}(\mathrm{m}^3) = \frac{\mathrm{mass(kg)}}{\mathrm{density}\ (\frac{\mathrm{kg}}{\mathrm{m}^3})} \tag{1}$$

For Nitrous Oxide stored as a self-pressurized liquid, the liquid density depends on the storage temperature because the tank contains both liquid and vapor in equilibrium. [5]

Table 3. *Typical Liquid N_2O Densities.*

Temperature (℃)	Liquid Density (kg/m^3)	Vapor Pressure (bar)
0 ℃	~900 kg/m^3	~30 bar
10 ℃	~840 kg/m^3	~44 bar
20 ℃	~760 kg/m^3	~51 bar
30 ℃	~620 kg/m^3	~71 bar

For most hybrid rocket preliminary calculations, engineers commonly use, Density at ~20 ℃ ≈ ~750–770 kg/m^3.

Using a typical value ≈ 760 kg/m^3: 1 litter $= \frac{760}{1000} = 0.76\ \ \mathrm{kg}$

$$V = \frac{4.8*0.76}{760} = 0.0048\ \ \mathrm{m}^3$$

R-32: Practical Diameter to Length Ratio of Tank Should Be Consider.

$$\mathrm{volume}\ \ (\mathrm{m}^3) = \mathrm{area}(\mathrm{m}^2) * \mathrm{lenght(m)} \tag{2}$$

In practical rocket design there's a well-established range that balances aerodynamics, structural strength, and stability. For most rockets, the length-to-diameter ratio (L/D) typically falls between: 10:1 to 20:1 → common for many sounding rockets and launch vehicles and Around 12:1 to 15:1 → often considered a good general-purpose design range. [6]

Let select L/D = 13

$$\mathrm{area}(\mathrm{m}^2) = \pi * \frac{\mathrm{D}^2}{4}(3) \tag{3}$$

$$\mathrm{V} = \pi * \frac{\mathrm{D}^2}{4} * 13\ \ \mathrm{D} \tag{4}$$

$\frac{4*0.0048\ \ \mathrm{m}^3}{13*3.14} = 0.00047 = \mathrm{D}*\ \mathrm{D}^2 = \mathrm{D}^3$; D = 0.0777498 meter, then L=1.011 meter.

R-33: tank volume should consider ullage space

For small rockets, ullage (the empty space in a tank) is mainly there to handle thermal expansion, sloshing, and to ensure propellant feed stays reliable. There isn't a single fixed number, but typical practice falls into a fairly narrow band. Common ullage percentages: ~2% to 5% of total tank volume → typical for many well-controlled designs and up to ~8–10% → used in simpler or less precisely controlled systems (including some small or experimental rockets). [6]

How to choose within that range: Lower end (~2–3%) Good temperature control, Pressurized systems with stable feed and more advanced designs where maximizing propellant volume matters. Middle (~3–5%) A safe, common default for small rockets and allows for modest thermal expansion and slosh. Higher (~5–10%) Simpler tanks without tight thermal control, larger temperature swings and Early-stage or experimental builds where safety margin is preferred over efficiency. [2]

Let used 4%, so updated volume $0.0048\ \ \mathrm{m}^3 +$ 4% of $0.0048\ \ \mathrm{m}^3 = 0.004992$

$\frac{4*0.004992\ \ \mathrm{m}^3}{13*3.14} = 0.0004892 = \mathrm{D}*\ \mathrm{D}^2 = \mathrm{D}^3$; D = 0.078794423 meter, then L=1.02433 meter.

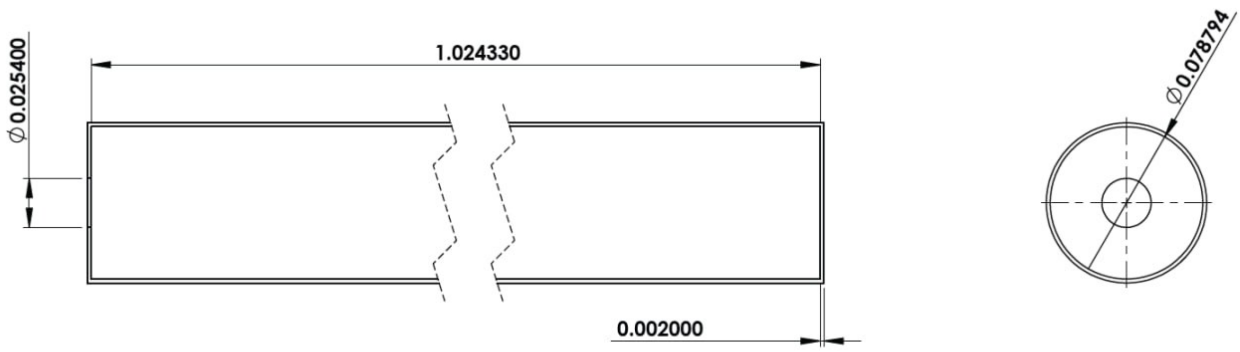

Figure 2. *Oxidizer tank dimension.*

$$\mathrm{Slenderness\ \ ratio} = \frac{\mathrm{Length}}{\mathrm{Diameter}} \tag{5}$$

S.r ≅ 13, means rocket under categories of slender rocket and needed to consider baffles.

Table 4. *Categories of Slenderness Ratio [7].*

No.	Slenderness ratio.	Names.
1	Below 5	Fat/stubby
2	5-10	Moderate
3	10-15	Slender
4	Above 15	Very slender and flexible

R-4: Propellant Filled Pressure in Tank, 51 Bar

R-41: The Ambient Temperature Should Not Exceed 20 ℃ While the Oxidizer Is Stored in The Tank.

The pressure–temperature relationship of nitrous oxide (N_2O) is important because it is a self-pressurizing oxidizer. As temperature increases, the vapor pressure rises rapidly.

Table 5. *Typical saturation-pressure values for nitrous oxide [8].*

Temperature (℃)	Vapor Pressure (bar)
-20	14.8
-10	20.0
0	28.6
10	39.0
20	50.7
25	57.3
30	64.7
35	73.0
36.4 (critical point)	72.5

R-42: Tank thickness must be calculated with enough safety factor

In order to find the thickness of the tank, For the cylindrical portion,

Hoop stress

$$\sigma_H = \frac{pd}{2t} \tag{6}$$

were

σ_H = Hoop Stress

P = Internal Pressure

d = Internal Diameter

t = Wall Thickness of The Cylinder

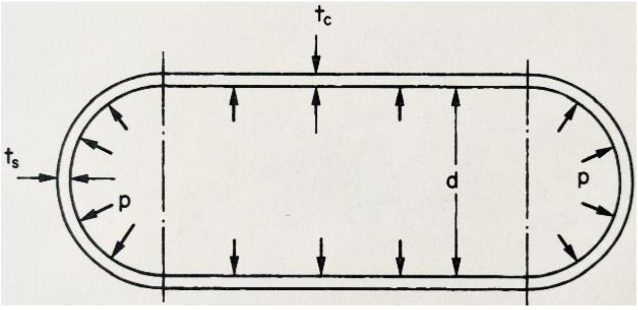

Figure 3. *Cross-Section of A Thin Cylinder with Hemispherical Ends.*

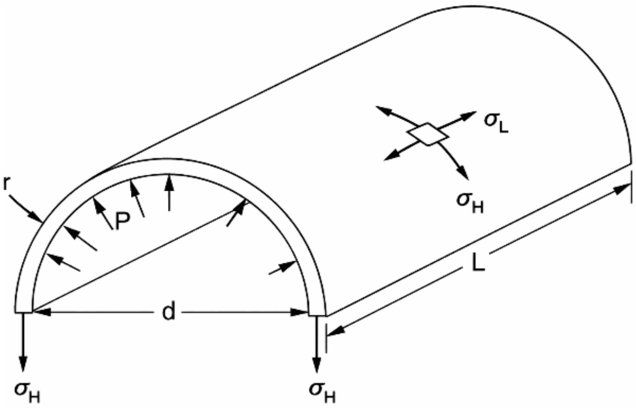

Figure 4. *Cross-Section of A Thin Cylinder.*

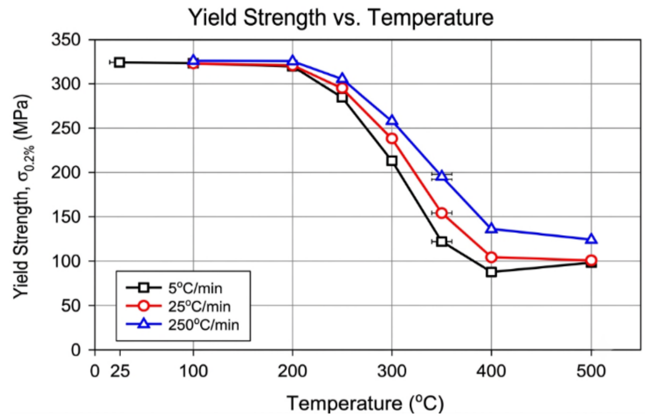

Figure 5. *Yield Strength (0.2% Offset) Of 6061-T651 After Prior Exposure at Different Heating Rates.*

Table 6. *Joint Efficiency of Different Weld Types [11].*

Types of weld	Joint efficiency, E		
	Full inspection requirement	**Spot inspection**	**Not inspection**
Butt joint as attained by double welding or by other means which will obtain the same quality of desolated weld material on the outside weld surface.	1.00	0.85	0.70
If a backing strip is used it should be removed after completion of welding	1.00	0.80	0.65
Single welding butt joint with backing strip remains in place after welding	0.90	0.80	0.85
Single welded butt joint without use of backing strip	0.60	-	0.60
Double full fillet lap joint	-	-	0.60
Single full fillet lab joint with plug welds	-	-	0.50
Single full fillet lab joint without plug welds	-	-	0.45

Table 7. *Yield Strength Values for Common Steels [12].*

Material	Yield strength	
	MPa	**Ksi (psi)**
Low alloy steel		
AISI 4140, Normalized at 870 ℃ (1600 ℉)	655	95 (95000)
AISI 4140, Annealed at 815 ℃ (1500 ℉)	414	60 (60000)
AISI 4140, Water quenched from 845 ℃ (1550 ℉) and tempered at 540 ℃ (1000 ℉)	986	143 (143000)
AISI 4340, Normalized at 870 ℃ (1600 ℉)	862	125 (125000)

325 MPa = $\frac{6\ \text{MPa} * 0.078794423\ \text{meter}}{2t}$ = 0.00072733 meter, and factor of safety 1.5 = 0.00109 meter (≅1 mm). In the case where welding mechanism is used to join an aluminum sheet, we use below equation. Joint efficiency varies with weld type. We choose the single full Not inspected/ single welded butt joint without use of backing strip and the joint efficiency 0.6.

$$\sigma H = \frac{pd}{2t\eta} \tag{7}$$

325 MPa = $\frac{6\ \text{MPa} * 0.078794423}{2*t*0.6}$, t = 0.00121221667 meter, With factor of safety 1.5 = 0.001818325 meter (≅2).

R-5: Vehicle Operation Altitude Domain, (0-20) Km

R-51 System should consider atmosphere temperature up to 20 km

As seen from below table the continuously decreases.

$$F = P * A \tag{8}$$

Were

$$F = force\ exert\ on\ the\ oxidizer$$

P= generated pressure (ullage)
A= service area of oxidizer (tank diameter)

$$A = \pi * \frac{d^2}{4} = 3.14 * \frac{(0.078794423)^{\ 2}}{4} = 0.004874\ m^2$$

$F = 5{,}100{,}000\ pascal * 0.004874m^2 = 24{,}857.4$ Newton

R-52 System consider air density up to 20 km

As seen from below table the air density due to altitude change not much, so it is reasonable and simplicity to assume the constant air density through vehicle fly time. System no need to include propellant management device (PMD) since not have the probability of microgravity environments.

Table 8. Source NRLMSIS Atmosphere Model.

Year	Month	Day	H (Km)	Lat	Lon	Air (Gm/Cm^3)	T (K)	T (℃)
2026	5	22	0	12.1	41.2	1.168E-03	299	25.88
2026	5	22	1	12.1	41.2	1.056E-03	294.7	21.55
2026	5	22	2	12.1	41.2	9.575E-04	289.3	16.15
2026	5	22	3	12.1	41.2	8.682E-04	283.3	10.15
2026	5	22	4	12.1	41.2	7.858E-04	277.1	3.95
2026	5	22	5	12.1	41.2	7.091E-04	271.3	-1.85
2026	5	22	6	12.1	41.2	6.374E-04	265.9	-7.25
2026	5	22	7	12.1	41.2	5.717E-04	260.5	-12.65
2026	5	22	8	12.1	41.2	5.124E-04	254.7	-18.45
2026	5	22	9	12.1	41.2	4.596E-04	248.1	-25.05
2026	5	22	10	12.1	41.2	4.128E-04	240.4	-32.75
2026	5	22	11	12.1	41.2	3.710E-04	231.5	-41.65
2026	5	22	12	12.1	41.2	3.325E-04	222.5	-50.65
2026	5	22	13	12.1	41.2	2.957E-04	214.1	-59.05
2026	5	22	14	12.1	41.2	2.603E-04	207.0	-66.15
2026	5	22	15	12.1	41.2	2.263E-04	201.7	-71.45
2026	5	22	16	12.1	41.2	1.941E-04	198.4	-74.75
2026	5	22	17	12.1	41.2	1.648E-04	196.9	-76.25
2026	5	22	18	12.1	41.2	1.387E-04	196.8	-76.35
2026	5	22	19	12.1	41.2	1.161E-04	198.0	-75.15
2026	5	22	20	12.1	41.2	9.688E-05	200.2	-72.95

Analysis of altitude-dependent atmospheric temperature and air density variations from 0 to 20 km under the environmental conditions corresponding to a specified date and year.

R-53 System consider gravitational force up to 20 km

As seen from below table the gravitational force due to altitude change not much, so it is reasonable and simplicity to assume the constant gravitational force through vehicle fly time.

Table 9. Gravitational Acceleration with Altitude (0-20 Km).

Standard Gravity	Mean Radius	Altitude (m)	Sum-1	Sum-2	Sum-3	Altitude Gravity
9.80665	6371000	0	6371000	1	1	9.80665
9.80665	6371000	1000	6372000	0.999843063	0.999686151	9.803572197
9.80665	6371000	2000	6373000	0.999686176	0.999372451	9.800495843
9.80665	6371000	3000	6374000	0.999529338	0.999058897	9.797420936
9.80665	6371000	4000	6375000	0.999372549	0.998745492	9.794347477
9.80665	6371000	5000	6376000	0.999215809	0.998432234	9.791275463
9.80665	6371000	6000	6377000	0.999059119	0.998119123	9.788204894

Standard Gravity	Mean Radius	Altitude (m)	Sum-1	Sum-2	Sum-3	Altitude Gravity
9.80665	6371000	7000	6378000	0.998902477	0.997806159	9.78513577
9.80665	6371000	8000	6379000	0.998745885	0.997493343	9.782068089
9.80665	6371000	9000	6380000	0.998589342	0.997180673	9.77900185
9.80665	6371000	10000	6381000	0.998432848	0.996868151	9.775937053
9.80665	6371000	11000	6382000	0.998276402	0.996555776	9.772873696
9.80665	6371000	12000	6383000	0.998120006	0.996243547	9.769811779
9.80665	6371000	13000	6384000	0.997963659	0.995931465	9.766751301
9.80665	6371000	14000	6385000	0.997807361	0.99561953	9.763692261
9.80665	6371000	15000	6386000	0.997651112	0.995307741	9.760634657
9.80665	6371000	16000	6387000	0.997494912	0.994996099	9.75757849
9.80665	6371000	17000	6388000	0.99733876	0.994684603	9.754523758
9.80665	6371000	18000	6389000	0.997182658	0.994373253	9.75147046
9.80665	6371000	19000	6390000	0.997026604	0.994062049	9.748418595
9.80665	6371000	20000	6391000	0.996870599	0.993750992	9.745368163

Variation in Earth's gravitational acceleration and the corresponding gravitational force over the altitude range of 0–20 km.

$$F = M * g \tag{9}$$

Were:
F = Force exerted in oxidizer due to gravity
M = Mass of propellant
g = acceleration gravity

Table 10. *Force Applied on Oxidizer (Propellant Due to Gravitational Force).*

Second	Propellant Consume Per Second	Current Propellant Mass	Gravitational	Force Due to Gravity
1	0.1824	3.648	9.8	35.7504
2	0.1824	3.4656	9.8	33.96288
3	0.1824	3.2832	9.8	32.17536
4	0.1824	3.1008	9.8	30.38784
5	0.1824	2.9184	9.8	28.60032
6	0.1824	2.736	9.8	26.8128
7	0.1824	2.5536	9.8	25.02528
8	0.1824	2.3712	9.8	23.23776
9	0.1824	2.1888	9.8	21.45024
10	0.1824	2.0064	9.8	19.66272
11	0.1824	1.824	9.8	17.8752
12	0.1824	1.6416	9.8	16.08768
13	0.1824	1.4592	9.8	14.30016
14	0.1824	1.2768	9.8	12.51264

Second	Propellant Consume Per Second	Current Propellant Mass	Gravitational	Force Due to Gravity
15	0.1824	1.0944	9.8	10.72512
16	0.1824	0.912	9.8	8.9376
17	0.1824	0.7296	9.8	7.15008
18	0.1824	0.5472	9.8	5.36256
19	0.1824	0.3648	9.8	3.57504
20	0.1824	0.1824	9.8	1.78752

Time-dependent gravitational force resulting from the oxidizer mass remaining in the tank throughout the burn duration.

Very small and Negligible compare to force ullage pressure, but total force push propellant down to feeding pipe = ullage pressure + gravitational ≅ *24,857.4 Newton*

R-6: Propellant Feeding Pipe Dimension 1 Inch Diameter And 0.1824 Kg/Sec Propellant Flow Rate.

R-7: Vertical Takeoff

Due to rocket alignment the vehicle can add additional advantage due to Gravity, and ullage pressure.

R-8: Powered Vehicle Flight Time (20 sec)

With the specified chamber pressure of 30 bar and the expected pressure losses in the feed system, the oxidizer tank pressure must remain sufficiently higher than 30 bar at the corresponding operating temperature. In the case of N_2O, the ambient temperature decreases significantly during flight. However, the oxidizer (N_2O) tank temperature changes much more slowly than the external air temperature because of the tank's thermal mass and the limited rate of heat transfer. [9]

R-9: vehicle deceleration deceleration up to 500 m/sec

Listed sample deceleration numbers at 250, 375, and 500 m/sec respectively for different mass (oxidizer) amount.

Table 11*. Generated Force Due to Vehicle Deceleration.*

Second	propellant consume per second	current propellant mass	deceleration 1	deceleration 2	deceleration 3	inertia F1	inertia F2	inertia F3
1	0.1824	3.648	250	375	500	912	1368	1824
2	0.1824	3.4656	250	375	500	866.4	1299.6	1732.8
3	0.1824	3.2832	250	375	500	820.8	1231.2	1641.6
4	0.1824	3.1008	250	375	500	775.2	1162.8	1550.4
5	0.1824	2.9184	250	375	500	729.6	1094.4	1459.2
6	0.1824	2.736	250	375	500	684	1026	1368
7	0.1824	2.5536	250	375	500	638.4	957.6	1276.8
8	0.1824	2.3712	250	375	500	592.8	889.2	1185.6
9	0.1824	2.1888	250	375	500	547.2	820.8	1094.4
10	0.1824	2.0064	250	375	500	501.6	752.4	1003.2
11	0.1824	1.824	250	375	500	456	684	912
12	0.1824	1.6416	250	375	500	410.4	615.6	820.8
13	0.1824	1.4592	250	375	500	364.8	547.2	729.6
14	0.1824	1.2768	250	375	500	319.2	478.8	638.4
15	0.1824	1.0944	250	375	500	273.6	410.4	547.2
16	0.1824	0.912	250	375	500	228	342	456
17	0.1824	0.7296	250	375	500	182.4	273.6	364.8
18	0.1824	0.5472	250	375	500	136.8	205.2	273.6

Second	propellant consume per second	current propellant mass	decelera-tion 1	decelera-tion 2	decelera-tion 3	inertia F1	inertia F2	inertia F3
19	0.1824	0.3648	250	375	500	91.2	136.8	182.4
20	0.1824	0.1824	250	375	500	45.6	68.4	91.2

Evaluation of the inertial forces produced under various deceleration scenarios and the corresponding oxidizer inventory remaining in the tank at each second of operation.

3. Result and Discussion

The analysis was conducted for an oxidizer tank containing 3.648 kg of nitrous oxide pressurized to 51 bars, with a vehicle operational altitude ranging from sea level to 20 km. Three primary forces acting on the oxidizer were evaluated: ullage pressure force, gravitational force, and inertial force resulting from oxidizer slosh during vehicle deceleration. The ullage pressure generated a nearly constant force of approximately 24,875.4 N acting on the oxidizer and gravitational force varied with altitude and oxidizer mass depletion, reaching a maximum value of 35.75 N at the beginning of the flight. In contrast, the inertial force associated with oxidizer slosh was evaluated for different vehicle deceleration conditions and reached a maximum value of approximately 1,824 N at a deceleration corresponding to a vehicle velocity of 500 m/s. Comparison of the calculated forces shows that the maximum slosh-induced inertial force is substantially lower than the opposing ullage pressure force. Specifically, the peak slosh force of approximately 1,824 N represents less than 8% of the 24,875.4 N force generated by the ullage pressure. Furthermore, even under increased deceleration conditions, the oxidizer mass considered in this study is insufficient to produce significant sloshing loads capable of adversely affecting vehicle stability or propellant feed performance. Therefore, throughout the entire flight profile and operational altitude range up to 20 km, the risk associated with oxidizer slosh remains negligible. Based on these results, the incorporation of internal baffles within the oxidizer tank is not required for the proposed hybrid rocket configuration.

Abbreviations

CEA	Chemical Equilibrium with Applications
N_2O	Hydroxyl-Terminated Polybutadiene
HTPB	Nitrous Oxide
NRLMSIS	Naval Research Laboratory Mass Spectrometer and Incoherent Scatter Radar
CP	Center of Pressure
CM	Center of Mass
PMD	Propellant Management Device
L/D	Length-To-Diameter Ratio

Author Contributions

Gedlu Solomon: Conceptualization, Methodology
Yishak Tamiru: Data curation, Resources

Conflicts of Interest

The authors declare no conflicts of interest**.**

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