COLORADO SCHOOL OF MINES
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NASA Human Lander Challenge

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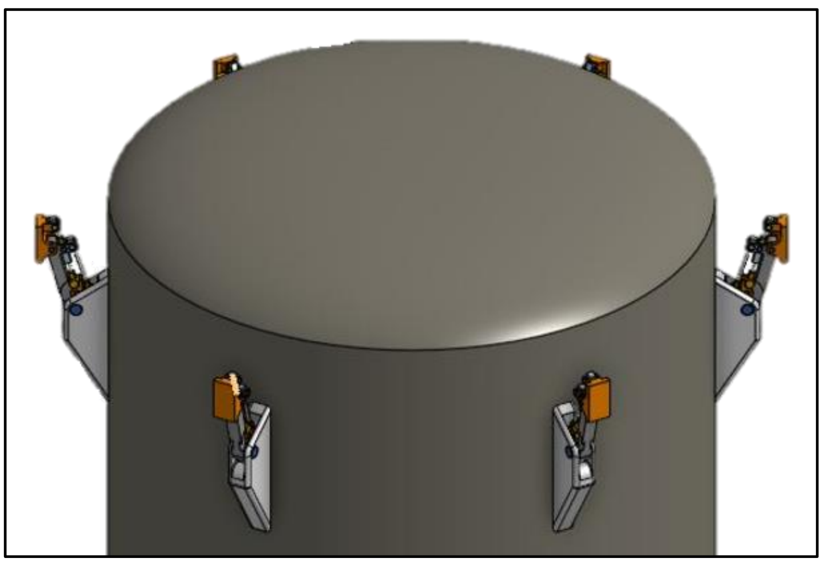

MAST – Modular Adaptive Separation Technology

28 May 2025

Colorado School of Mines – MAST (Modular Adaptive Separation Technology)

Theme Category, Major Objectives, & Technical Approach

- Advanced Structural Supports for Heat Reduction
- Goals
  - Minimize heat transfer
  - Adapt to low-load conditions
  - Withstand launch forces, vibrations, and lunar landing impulse
  - Work in cis-lunar and lunar environments
- Technical Approach
  - Utilize existing technologies where available
  - Verify design through SolidWorks Simulator
  - Using finite element, conduct static load and thermal analyses

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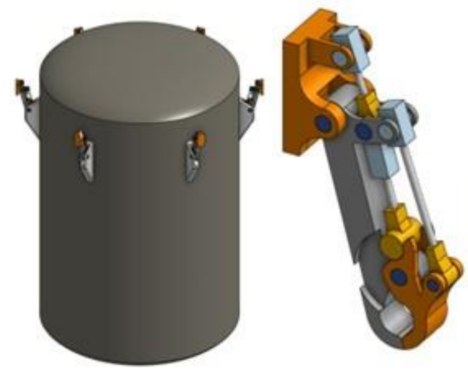

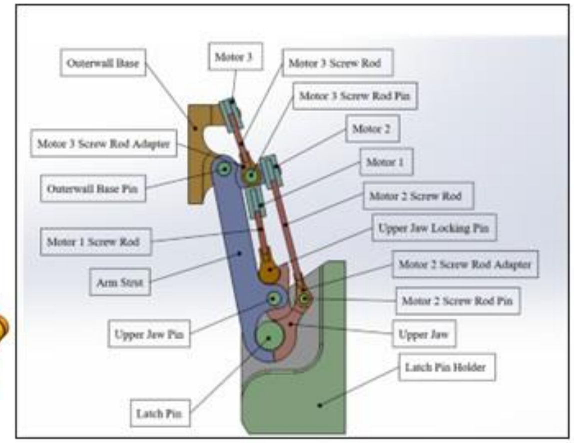

Key Design Details & Innovations of the Concept

Design Details

- Triple Actuator System
- Welds to fuselage and skirt
- Jaws unclamp to remove contact

Innovations

- Decoupling of conductive elements
- Modular in design
- Removes medium for heat transfer
- Reduces propellant boiloff

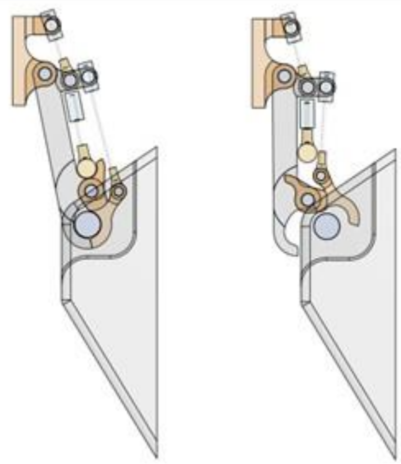

Figure: Jaws closed at launch (left), Jaws opened post launch (right)

Summary of Schedule & Costs

- Calculated Costs: Materials, Components, Manufacture and Assembly, Safety, Launch
- Uncertainty for upper bound estimates
- Lunar landing cost: $288,299,672
- With uncertainty: $358,927,912.80
- LEO launch: $1,936,472
- With uncertainty: $2,998,912.80

Phase	Estimated Time (Months)
Planning	4.5
Testing and Iteration	9.5
Manufacturing for Assembly	4
Assembly	3.5
Integration with Spacecraft	3
Safety Checks	2
Total (Months)	26.5

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1. Executive Summary

For the 2025 Human Lander Challenge, the Colorado School of Mines team, PESTO, has chosen to target advanced cryogenic supports. PESTO is proposing their novel solution Modular Adaptive Separation Technology (MAST). MAST aims to reduce the conductive heat transfer moving from the fuselage of a spacecraft into the propellant tanks through the utilization of a series of decoupling supports in conjunction with a small number of static welded supports depending on tank size. MAST supports can be selectively coupled or decoupled from the propellant tank to meet the demands of each mission phase, creating breaks in the pathways through which heat transfer can occur, while maintaining the necessary strength for the most structurally demanding portions of the mission. The result of this solution is a drastic reduction of heat transfer into the propellant, from a figure on the order of kilowatts down to 28 Watts per support.

2. Project Description

I. Problem Introduction

Cryogenic fluid boiloff due to excessive heating is a major issue that has plagued the aerospace industry for almost as long as it has existed. Cryogenic propellants such as hydrogen, methane, and oxygen are an extremely desirable fuel for space flight due to the extraordinary specific impulse. Due to the extremely low temperature these fluids must be stored at, they are highly susceptible to even small amounts of heat leaking in. Small heat additions cause the fluids to vaporize, rapidly expanding, requiring regular venting for safety. Boiloff is a critical issue if left unaddressed, reducing the amount of consumable propellant available in addition to potentially posing significant safety risks to crew members on manned missions. Conservation of available propellant is extremely important for long duration missions [1]. The longer a thermal system is in space, the closer it becomes to achieving thermal equilibrium which, for the purpose of maintaining cryogenic temperatures for a propellant tank, is undesirable. PESTO is faced with improving upon existing cryogenic tank support systems aiming to maintain mission safety standards while substantially reducing or mitigating heat transfer to prevent boiloff.

II. Solution Overview

In solving this issue, PESTO is proposing their novel solution Modular Adaptive Separation Technology (MAST). MAST can drastically reduce heat transfer to cryogenic propellant tanks by decoupling a selected number of supports, creating a thermal break where conductive heat transfer cannot occur. MAST works with a system of dynamic supports, evenly spaced with symmetry around the top and base of a propellant tank. These supports bridge the gap between the fuselage and propellant tanks. Individual supports can be coupled or decoupled to adhere to the support needs of each mission phase while minimizing pathways in which heat transfer can occur. MAST will operate in one of two phases: high-load and low-load. MAST is modular, as such, mission parameters can dictate the number of supports engaged in high-load phase. High-load phase is during periods of high acceleration, such as launch, landing, or orbital maneuvering. Low-load phase is designed to be active more frequently, during the long periods of orbit or low-acceleration.

During the low-load phase most of the supports will decouple, reducing the conductive heat transfer into the propellant.

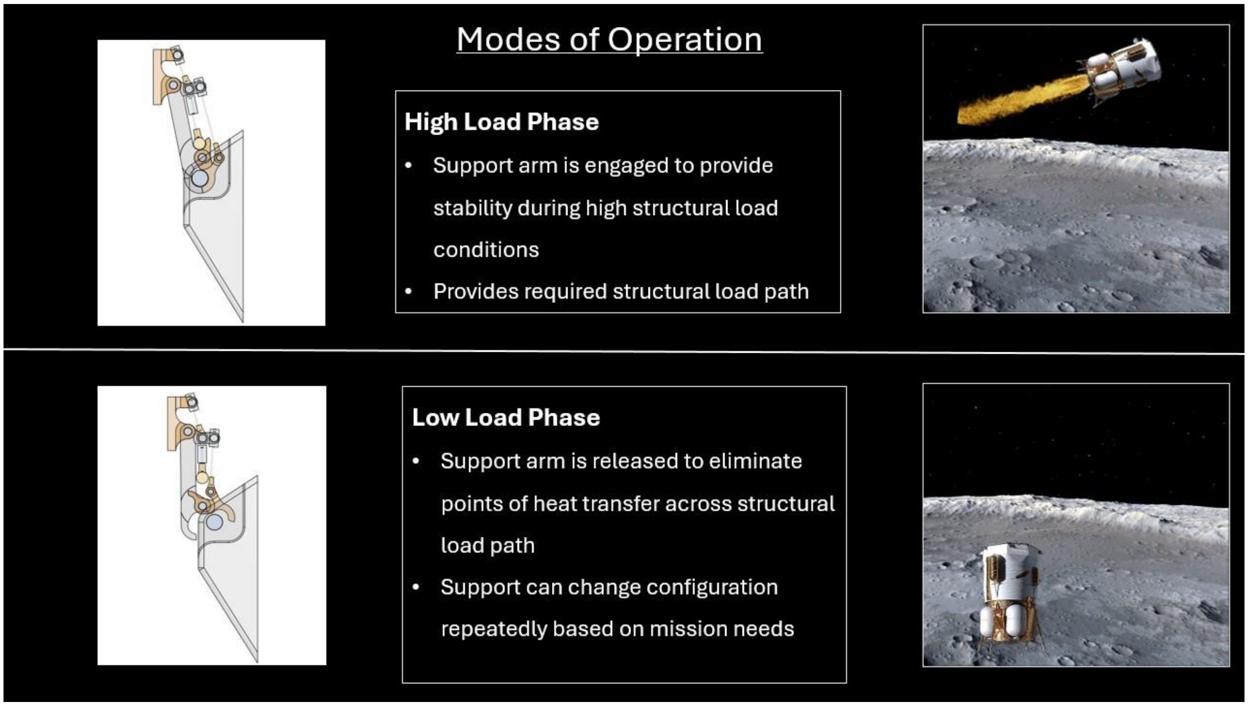


Figure 1 – *Operational viewpoint.*

Figure 2 below shows the arrangement of the MAST supports around a propellant tank, minus the static welded supports that work in conjunction to maintain alignment while the MAST supports are in their low load configuration.

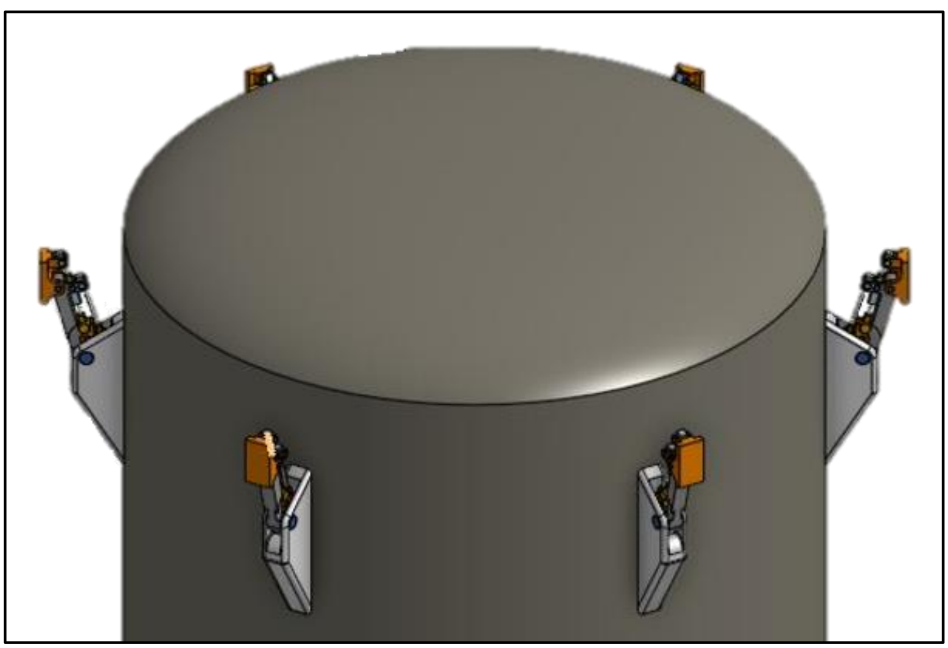


Figure 2 – *Arrangement of MAST supports around a propellant tank.*

MAST activates by retracting the upper jaw locking pin, allowing Motor 2 Screw Rod to retract. This disengages the upper jaw. Lastly, motor 3 extends its corresponding screw rod, shifting the lower jaw to the left and removing all contacts. The process reverses for reattachment.

MAST's dimensions are 55.4 x 29.9 x 10 cm, weighing 23.8 kg. See Appendix 4 for complete dimensions.

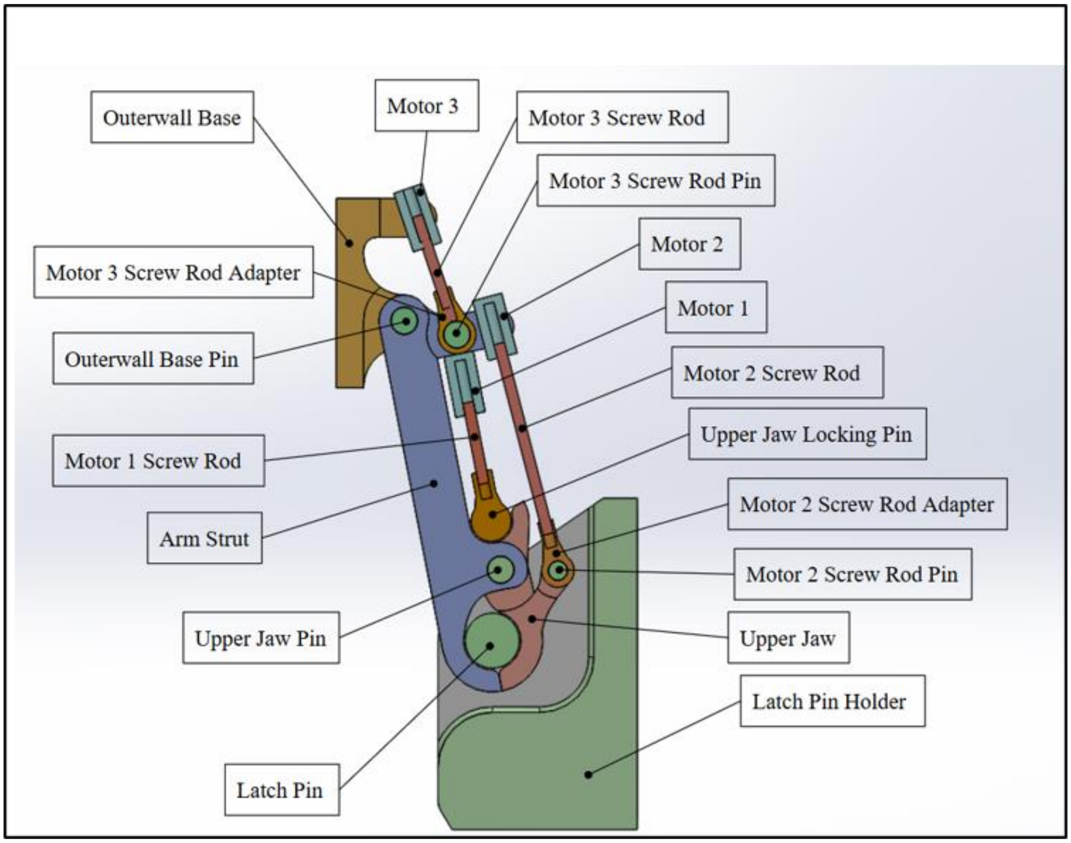


Figure 3 – *Cross section of engaged MAST support.*

III. Requirements & Verification

Table 1 – *High-level requirements.*

Type	Level	Req	Requirement	Verification Method	Verification Rationale
High Level	**1**	**HL1**	The design shall have minimal barriers to NASA adoption.	Testing and Analysis	Concept TRL must be maximized for ease of adoption. The solution must also not interfere with boundary systems like tank skirt and fuselage.
High Level	**1**	**HL2**	The design shall survive in a space environment.	Testing and Analysis	The solution needs to survive extreme temperature gradients, micro gravity, vacuum, and radiation. This can be

					verified through component tests and thermal analysis.
High Level	**1**	**HL3**	The design shall be fieldable in 3 – 5 years (Artemis III).	Analysis	Material components need to come from verified existing technologies, and the proposed schedule needs to be accurate.
High Level	**1**	**HL4**	The design shall not pose additional risks to the crew.	Testing and Analysis	The design cannot include anything that imposes mission or life-threatening risks to the crew. A risk analysis table is used for verification.
High Level	**1**	**HL5**	The design shall survive launch and landing loads.	Testing and Analysis	The solution needs to withstand certain vibrations, aggressive landing forces, and varying challenges associated with launching the system. This will be done through Finite Element simulation.
High Level	**1**	**HL6**	The design shall survive a mission duration of multiple months.	Testing and Analysis	The design requires no maintenance and utilizes materials suited to last extended durations in a space environment.

To verify whether the mechanical system is suitable for launch loads, PESTO utilized SolidWorks software. The parts in the mechanical system were designed in SolidWorks and will be tested under launch loads through a finite element model. Available load data is taken from a mass velocity curve [Fig. 4 below]. Similarly, for the purposes of analyzing heat transfer PESTO will also use SolidWorks simulation.

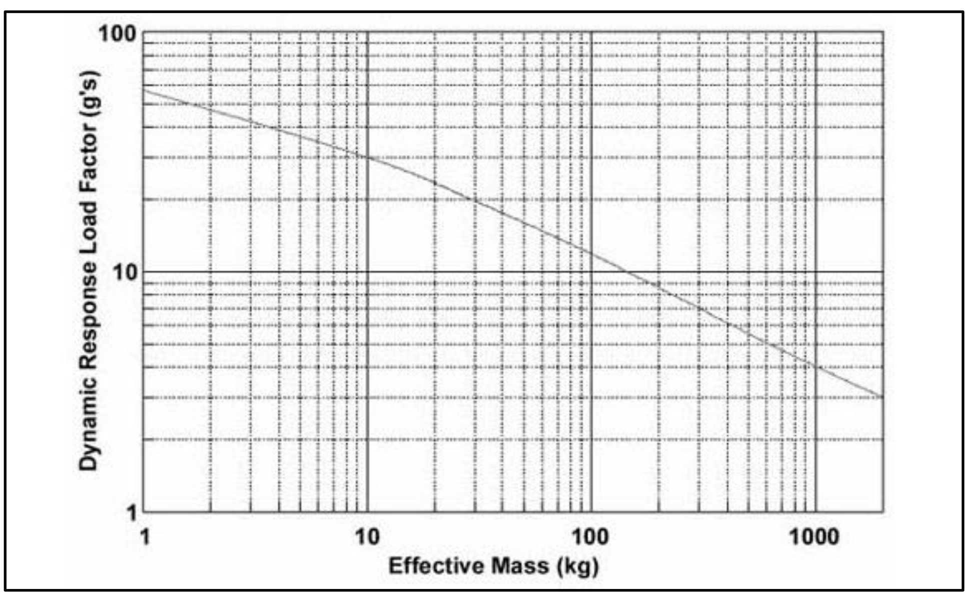


Figure 4 – *Typical mass acceleration curve for vibrations.*

IV. Alterations from Proposal

Since the proposal, the name of the technology, Modular Adaptive Support Technology has been updated to better reflect its purpose, MAST is now Modular Adaptive Separation Technology.

Following the proposal, a mistake was discovered in the finite element model which significantly affected the stress present in the assembly. This was fixed in this paper making the stresses and factor of safety of the system more accurate. Along with this, the size and mass of the overall system has changed and therefore the values of heat transfer and vaporization have been updated.

Several cuts and fillets have been added to the model to optimize mass and remove stress concentration locations. This retains performance and improves costs.

Cost estimations have been adjusted to reflect new model properties and feedback.

V. Innovations

MAST addresses the challenges associated with cryogenics boil-off reduction and thermal insulation, essentials for long-term and deep space missions. This technology aligns with NASA technology taxonomy areas [TX14.2] for thermal control and components, [TX13.1] for propellant management and storage, and [TX12.2] for innovative and adaptive mechanical systems.

MAST provides a uniquely innovative and scalable solution enabling the simple integration of existing technology into a new functional component, capable of significantly reducing heat transfer into cryogenic stores allowing for extended mission times and making deep space exploration possible.

3. Engineering Analysis

I. Simulation Assumptions

Simulation parameters involve considerations for launch, landing, and thermal cases. Launch acts as an envelope case for stress testing. Three liquid fuels were considered: Oxygen, Methane, and Hydrogen. Oxygen was used in structural simulations as it has the greatest average density. Tank size varies depending on whether a large launch system or a lunar lander module is considered.

SolidWorks Simulation packages were used to conduct thermal and structural analysis. The Finite Element Analysis capabilities of SWS will be used for each of the three test cases. The objective of the FEA tests was to maintain a minimum factor of safety of 1.5 for the mechanical loading cases, and to mitigate heat transfer in the thermal loading. From there, the mass and size of MAST's profile was reduced to more closely align with mass requirements.

The only source of heat assumed in the system is directed from solar radiation. This incoming heat is assumed to be always direct, with no eclipse period, despite much of the fuselage being out of direct sunlight. This would suggest that the values calculated for conductive heat transfer are done in a worst-case scenario and are very conservative, where the values are likely substantially reduced in practice through reflective coatings on the fuselage, and in periods of eclipse where solar radiation flux is at a minimum.

The titanium alloy Ti-5Al-2.5Sn [Table 2, below] was selected, given its previous use in aerospace applications due to its low density relative to its high-performance and low thermal conductivity characteristics [2]. Outside of the actuators and electronics, every element of the assembly was given the titanium alloys material properties for loading and thermal analysis.

Table 2 – *Ti-5Al-2.5Sn material properties.*

Titanium Ti-5Al-2.5Sn - Material Properties	
Tensile Strength, Ultimate	861 Mpa
Tensile Strength, Yield	827 Mpa
Modulus of Elasticity	110-125 GPa
Density	4480 kg/m^3
Thermal Conductivity	7.8 W/m-K
Specific Heat Capacity	0.53 J/g-°C

II. Mechanical Analysis

Launch analysis was performed considering the usage of a Falcon Heavy launch system with maximum tank dimensions. Based on competition guidelines, the cylindrical tank is six meters in diameter and 10 meters in height, with 10% domes for the end caps. Using liquid Oxygen as a test fuel, the gravitational force acting on the latch pin can be determined. The mass of the tank and MAST were considered negligible compared to the propellant.

Current launch systems generate ~3 g's of acceleration upon launch. Using launch acceleration, the force acting on the latch pin can be multiplied by a factor of three to find the launch force used in the simulation. At the current tank parameters, the launch force is ~6.96 million Newtons acting on all supports and the bottom of the fuselage in the downward direction.

Table 3 – *Thrust calculator.*

Calculator							
Dia (m)	6	**Height (m)**	10	**Fuel**	Oxygen	**Domes:**	10%
				Density:	1141		
Volume:	207.3449	**m^3**					
Mass:	236580.6	**kg**					
Launch Acceleration (G)			3				
Launch Force (N)			6,962,566.37				

Using a mass-acceleration curve (MAC) [Fig. 4], the response loading factor is determined. In contrast to the large loading case for thrust, the response loading factor is increased with smaller masses. For launch, only about 1 g of acceleration will occur since the tank is so large. This occurs on all three coordinate axes. That was used, in conjunction with launch loads, to adequately account for the maximum conditions the support will experience at launch.

The SolidWorks model used fixed geometry on the outer wall base and roller geometry on the latch pin holder; it can be assumed the joints hold. Additionally, each actuator was designed to experience minimal loads, so that the arm strut, upper jaw, and latch pin/latch pin holder bear all the loads. Consequently, each actuator was given a fixed geometry to ensure the simulation does not account for them as load bearing. The mesh created used bonded geometry, aside from the upper and lower jaws, and the latch pin. These were given contact relations.

For loading, 1 g of vibrational loading was applied in each direction. There was also an additional 3 g's of thrust acting downward. From the force of the tank, it was assumed that half of the thrust would be supported by the base support at the bottom of the tank. Twelve support structures were placed around the propellant tank, split into two layers of six supports. On the latch pin holder face, 290 kilonewtons of force was estimated on the individual support.

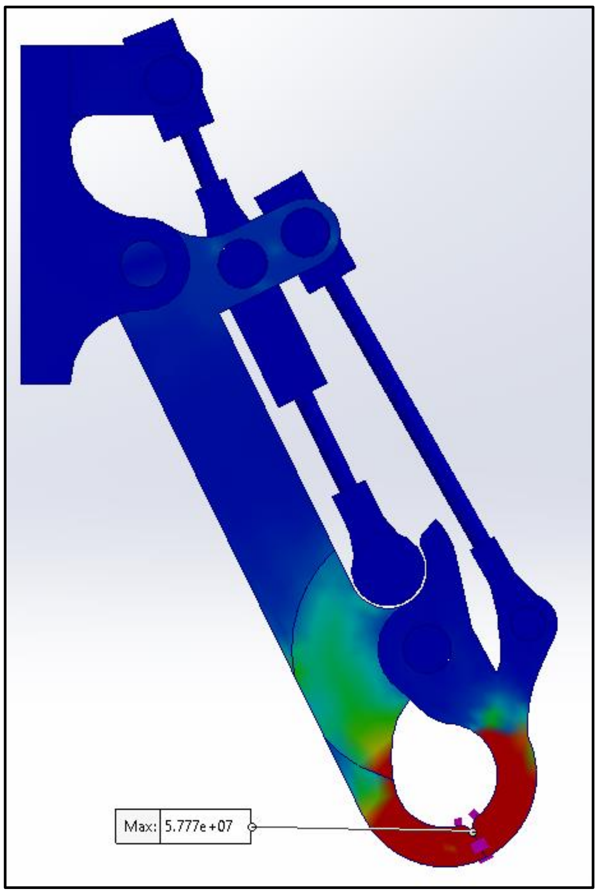


Figure 5 – *SolidWorks launch stress simulation results.*

Launch results place the greatest amount of stress where it was to be expected: on the bottom of the jaws. The maximum von Mises stress was 57.8 MPa on the face contacting the latch pin. This is where the lowest Factor of Safety was recorded, at 14.31.

A drop test in SolidWorks acted as a benchmark for landing on the moon and supporting a lunar lander. At one-sixth of the gravity of Earth, lunar landers typically land at ~3.1 feet per second [3]. The drop test rendered a factor of safety 10.3 showing promising results for lunar applications.

III. Thermal Analysis – Heat Transfer

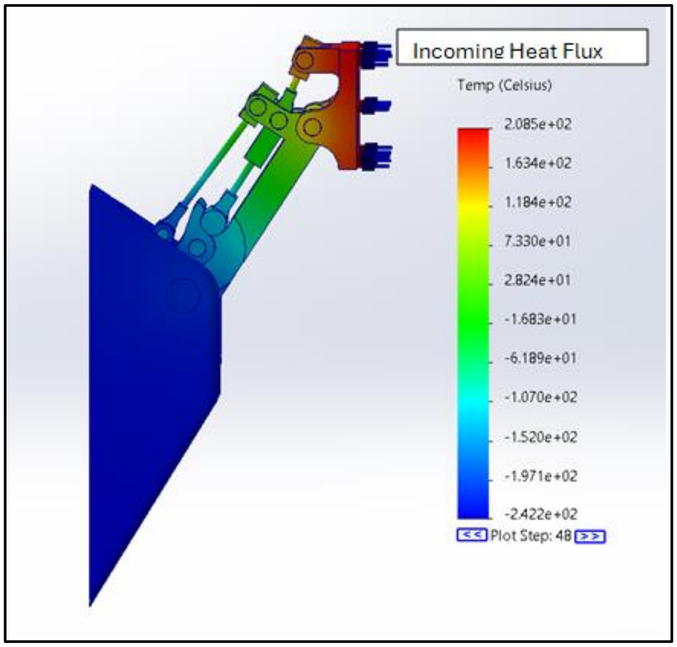
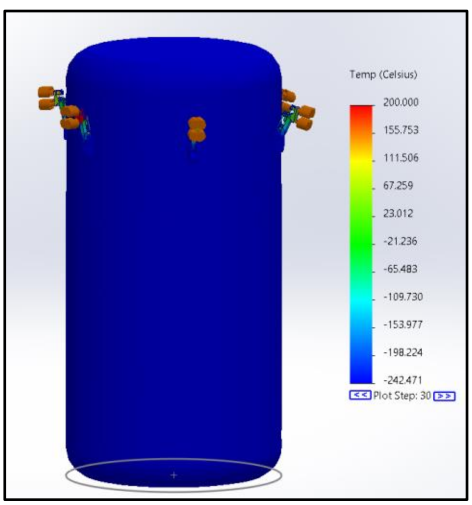
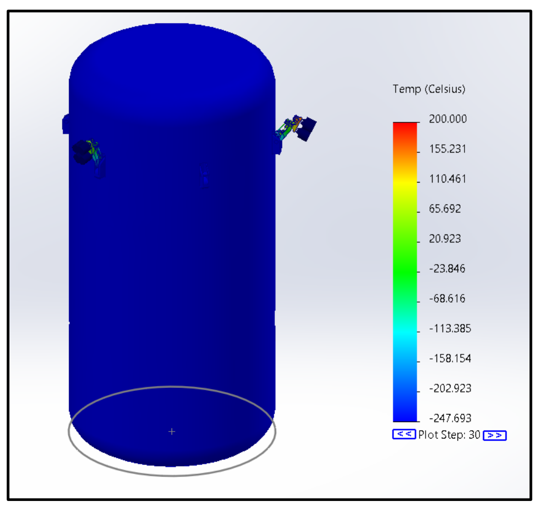


Figure 6 – *SolidWorks heat transfer simulation results.*

Figure 6 above is the resulting temperature plot after 1 day of constant heat flux from the given initial temperature and incoming flux conditions. It shows that the used titanium alloy has a small thermal conductivity, effective in transferring minimal heat through its body.





Figures 7 (left), 8 (right) – *SolidWorks heat transfer simulation results for full assembly.*

For the full assembly simulations, the analysis was conducted with a timeframe of 30 days with an increment of 1 day. Detaching half of the supports resulted in a decrease in the final lowest temperature of 5 degrees. For visibility, the maximum temperature shown on the plot was limited to 200 degrees Celsius.

Given the stated heat transfer rate of kilowatts during the HuLC Q&A, a comparison was made using the heat transfer values obtained through simulation to calculate boiloff rates for conventional supports and for MAST. To calculate boiloff rates, the time derivative of the equation for phase change was used, where m is the mass quantity of propellant and L is the latent heat of vaporization.

$$\dot{Q} = mL \quad [1]$$

For the comparison calculation, a tank size of 1 by 3 meters was selected, with hydrogen as the fuel. Based on this, an input of 2500 watts was assumed for conventional support structures, and the boiloff was calculated to be 6.85 g/s. For MAST, the calculated heat flux based on the simulations was approximately 28 watts, giving a boiloff rate of 0.378 g/s.

IV. Analysis Discussion

Based on the simulation results in static, vibration, and thermal aspects, the design fulfills the requirements shown in Table 4 below.

Table 4 – *Validation of requirements.*

Requirement	Simulation Result	Requirement Met
PR1 Prevents substantial order of kilowatts from entering tank.	Latch pin holder maintained steady state temperature of -242.2° C. Estimated around 28W transfers into the tank at the worst conditions.	YES
PR2 Support tank during high acceleration(5G).	Factor of safety for assumed acceleration of 5G downward into addition to vibrational acceleration of 3G for a total of 8G downward resulted in FoS of 14.34.	YES
PR3 Survive vibration loads(3G).	Stress analysis included the 3G loading in other two directions perpendicular to downward loading. FoS was 14.34	YES
PR4 CTE for fuselage and base roughly equivalent.	Aluminum 2024 –T4 CTE: 24.7μm/m-° C Ti-5Al-2.5Sn CTE: 9.5 μm/m-° C	YES
PR6 Support tank against 116 kN of downward force.	Simulation found FoS of 14.34 for the worst condition with the highest acceleration.	YES
ST2 Single support has mass less than 25kg.	Individual support met the 25kg limit at ~24kg.	YES
ST3 Mechanical FoS no less than 1.5 anywhere on support.	Simulation found FoS of 10 for the worst condition with the highest acceleration.	YES

V. Limitations

This analysis only considers the major structural components which provide support to the cryogenic propellant tanks within a rocket. This does not include additional components and points of which contact the tanks, such as sensors, turbopumps, valves, and other plumbing through which heat would transfer. This analysis only considers heat transfer through conduction, disregarding radiation and convection. It is likely that a substantial amount of heat would be introduced to the system by these means that were not considered in this evaluation.

4. Implementation & Project Management

I. Bill of Materials

Shoulder bolts will replace the pins shown in the assembly, allowing for free movement of the actuators, while restricting translation, like a hinge. In total, the standard MAST arrangement calls for 12 total arm assemblies, so each item in the BoM can be multiplied by 12 in accounting for the entire system.

Table 5 – *Bill of materials (BoM).*

ITEM NO.	PART	QTY.
1	OUTERWALL_BASE	1
2	UPPER_JAW	1
3	ARM_STRUT	1
4	Linear Solenoid 1	1
5	Linear Solenoid Fasteners	4
6	Shoulder Bolt	6
7	UPPER_JAW_LOCKING_PIN	1
8	0.4" Stroke Actuator	1
9	2" Linear Actuator	1

II. Controls Implementation

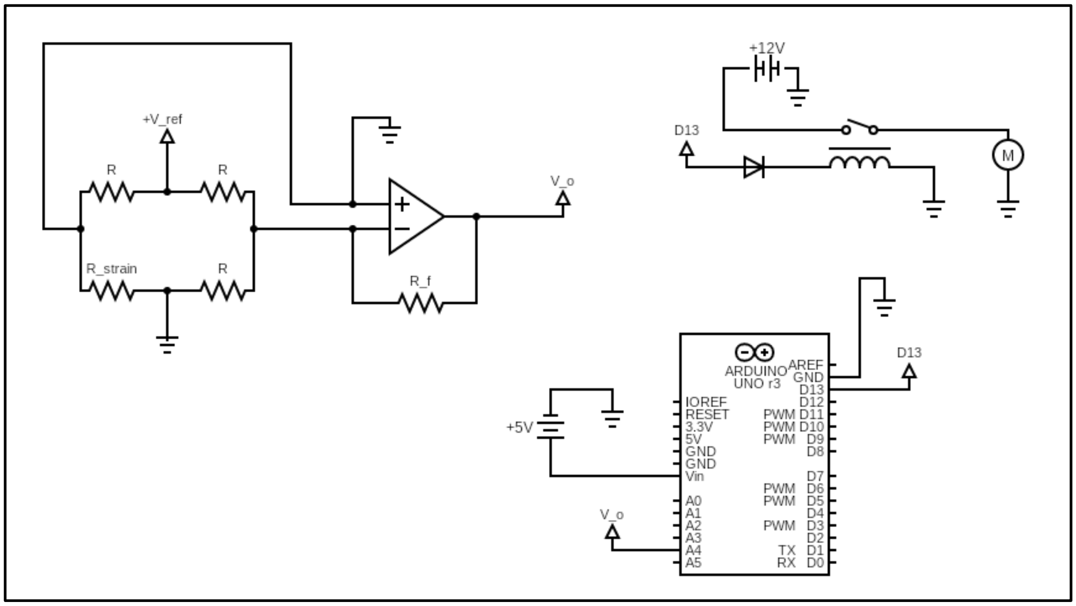


Figure 9 – *Prototype circuit diagram*.

Figure 9 above shows the general simplified circuit layout that will be used during prototype testing. The circuits use a strain gauge built into a Wheatstone configuration to accurately measure the small changes in the strain gauge resistance. Such strain gauge sensors are common in the current sensor market, such as the SGT-4/1000 series on DigiKey (Manufacturer Part Number: SGT-4/1000-FB13) where the Wheatstone bridge is already built into the transducer. Following the strain gauge circuitry, an amplifier circuit will be utilized to achieve greater response to resistance changes in the strain gauge. The gain for this amplifier circuit can be adjusted with the Feeback resistor to make smaller changes in the strain gauge resistance pronounced but not enough to fully saturate the amplifier. The microcontroller in the bottom right of the figure uses net labels to highlight that the output from the amplifier circuit is fed into the analog input for additional processing and logic control. Then a digital output pin is used to control a relay which will allow a larger 12V supply to power/control the motors in the linear actuators.

III. NASA Implementation Plan

The itemized schedule uses similar manufacturing periods for more complex objects like CubeSats to estimate the necessary time for project completion. Schedule estimation is just under two years for full implementation, fitting well within the 3–5-year implementation window.

Table 6 – *Summary of time estimates.*

Considering the increased cost of production and implementation, the additional performance gained will help offset these costs. These performance gains include preserving propellants and minimizing refueling missions. This is necessary for long-term missions with cryogenic propellants.

IV. NASA Cost of Implementation

Estimates based on NASA's planning guide and launch costs have been made to facilitate implementation of MAST at full scale [4]. Cost basis has been through multiple categories, including material costs, manufacturing, testing, and launch. Each of these carries an associated uncertainty, to give a better understanding of the upper bound estimates. Both Lunar landings and LEO launches have been calculated to better understand the flexibility of MAST and how much launch trajectory impacts cost [5] [6]. Guidelines dictate that the solution must be applicable to both orbiting refueling stations and lunar operations. Testing includes determining stress limits, dropping tests for landing, and large accelerations in a vacuum. Calculations considered the manufacture and integration of 12 total supports. A large-scale tank will use two levels of six supports.

Table 7 – *Summary of costs.*

Items	Lunar Landing Cost	Lunar (UC)	LEO Launch Cost	LEO (UC)
Resource	Cost	Uncertainty Cost	Cost	Uncertainty Cost
Material	$216,672.00	$246,412.80	$216,672.00	$246,412.80
Manufacture	$53,000.00	$70,000.00	$53,000.00	$70,000.00
Integrate and Test	$2,430,000.00	$1,611,500.00	$810,000.00	$1,611,500.00
Launch	$285,600,000.00	$357,000,000.00	$856,800.00	$1,071,000.00
Total	**$288,299,672.00**	**$358,927,912.80**	**$1,936,472.00**	**$2,998,912.80**

Table 8 – *Comprehensive cost calculations.*

Resource	Cost Type	Amount	Cost Per Unit	Cost	Uncertainty	Cost UC
Ti-5Al-2.5Sn	Per kg	571.2	$60.00	$34,272.00	0.15	$39,412.80
Stepper Motors	Per Unit	36	$5,000.00	$180,000.00	0.15	$207,000.00
Circuitry	Per Unit	12	$200.00	$2,400.00	0.10	$2,640.00
Manufacturing	Per Hour	160	$200.00	$32,000.00	0.40	$44,800.00
Testing		1	$800,000.00	$800,000.00	1.00	$1,600,000.00
Assembly	Per Hour	60	$350.00	$21,000.00	0.20	$25,200.00
Safety Checks	Per Check	2	$5,000.00	$10,000.00	0.15	$11,500.00
Launch Cost (Lunar)	Per kg	285.6	$1,000,000.00	$285,600,000.00	0.25	$357,000,000.00
Total			Total	$286,679,672.00	Total (UC)	$358,930,552.80

V. Risk Assessment

Table 9 – *Risk assessment consequence decoder.*

	Likelihood	Cost	Schedule	Technical
1	1%-10%	<1% Program Budget	1 Week Delay	Minimal Impact
2	11%-30%	1 to 2.5% Budget	2 Week Delay	Minor Impact on performance or mission
3	31%-60%	2.5 to 5% Budget	Multi-week delay to delivery	Moderate impact to performace, low impact on mission
4	61%-80%	5 to 10% Budget	Month long delay to delivery	Large impact to performance, moderate impact to mission
5	81%-99%	>10% Budget	Multi-month long delay to delivery	Massive impact to performance, almost certain mission loss

Table 10 – *Risk assessment matrix.*

Risk	Risk Statement (if/then)	Cost Consequence Rating	Technical Consequence	Schedule Consequence Rating	Likelihood Rating	Status	Mitigation Plan
1	If the structure is damaged during the mission, then it must be repaired.	5	5	3	1	M	Thoroughly analyze and test the unit to qualification load levels to mitigate risk of failure
2	If the prototype is not complete, then there will be less confidence in the design working.	1	1	1	3	A	Have deadlines for part ordering and construction.
3	If the spacing between the skirt and fuselage is too small, then the design will need to be altered.	3	2	4	2	M	Research clearances based on current tank fixture designs and implement changes before fabrication.
4	If the design is too heavy, then it will not be approved for space travel and cost will increase.	2	2	3	3	M	Optimize support structure for mass as well as area.
5	If the structure cannot withstand launch forces, then an alternative solution must be used.	2	5	3	2	M	Use Simulation tools and experiments to replicate practical launch environment.
6	If the structure allows a large amount of heat transfer, then heat will need to be removed from the cryogen tanks.	2	4	2	3	M	Multiple configurations of the design will be tested, to minimize heat transferred.
7	If the design is too expensive or complicated to manufacture, then the design will need to be changed.	3	2	3	2	W	Use a decision matrix balancing both structural complexity and strength.

Table 11 - *Analysis of likelihood and consequences by risk.*

Likelihood Rating						
	5					
	4					
	3	2			3 6	
	2			7		5
	1			4		1
		1	2	3	4	5
		Consequence Rating				

Risks associated with development, implementation, and operation were assessed. Assessment views various roadblocks or concerns that could arise, through which flaws and fatal errors are explored. Risks are assigned likeliness ratings, as well as consequence ratings for each of the three consequence categories. Severe risks for mitigation are labeled 'M' and use mitigation plans to decrease likelihood and consequence. For risks labelled as 'A,' it indicates that the risk falls under the 'accepted' category. For risks labeled as 'W,' it indicates that the risk falls under the 'watch' category. Through mitigation and careful assessment, all risks will be driven to acceptable levels or retired.

Risk 1 may be mitigated through complete testing; if the structure survives launch loads, then it will survive the remainder of a mission. Launch is considered the most stress intensive phase. Risk 3 concerns the bonds between the support structure, the fuselage, and tank skirt. MAST was designed to be comparable in width to arm struts, verifying its design. Risk 5 was mitigated through stress analysis. Lastly, Risk 6 was mitigated through thermal analysis compared to standard fixed supports.

VI. Testing Plans

With FEA showing the load bearing and heat transfer capabilities of the system, the next step is to manufacture a scale model using the intended material. This allows electronics and controls to be tested alongside real-world stress and thermal testing.

To ensure safety, the model needs to be able to withstand the scaled stresses of launch without dislodging prematurely. The test will utilize two fixed points/walls which vibrate to simulate the launch loads. The FEA indicates that the connecting joints are the weakest members. Forces on these members will be simulated using this vibration for 15 minutes since extreme launch loads typically occur for 5 to 10 minutes [7]. The system will then be inspected for imprints, damage, and material strain, all of which will be recorded. 20 tests shall be conducted in this manner to identify risks over time, and the strength of the solution.

If the mechanism does not exhibit signs of fatigue or failure, an additional test will be conducted to increase the load until the mechanism fails. Failure of the mechanism is described as permanent disfigurement of material, a latch opening, or a latch no longer being able to operate. This test will be conducted in 15-minute intervals where the launch load is continuously increasing. This test will operate until the failure described above and continue to collect thermal data.

5. Summary

The MAST solution was designed to meet the intended stress tolerance requirements and thermal mitigation goals. Current skirt designs maintain a lot of contact between the vessel airframe and the tank. This makes these tanks susceptible to heat transfer and cryogenic boil-off. MAST aims to reduce heat transfer by decreasing the surface area in contact with cryogenic propellant tanks. MAST is adaptable, conforming to the needs of the mission and possible future missions. This innovative and novel solution will advance efforts towards building more sustainable lunar and cislunar operations, supporting Moon-to-Mars architecture for the future of the Artemis program and programs that succeed it. Minimizing size, cost, and considering modularity to fit current spacecraft, our design provides invaluable gains in heat transfer mitigation and mission longevity by reducing heat transfer by a projected X%. It satisfies competition guidelines and fits within NASA's goals for the future in supporting current, ongoing, and planned future cislunar operations, lunar settlements, and Moon-to-Mars architecture.

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Appendix

Appendix 1 – SWS Static Mesh Parameters

Mesh Details

Study name	Launch (-Default-)
DetailsMesh type	Solid Mesh
Mesher Used	Blended curvature-based mesh
Jacobian points for High quality mesh	16 points
Max Element Size	11 mm
Min Element Size	1.22423 mm
Mesh quality	High
Total nodes	76134
Total elements	48107
Maximum Aspect Ratio	242.88
Percentage of elements with Aspect Ratio < 3	99.1
Percentage of elements with Aspect Ratio > 10	0.116
Percentage of distorted elements	0
Number of distorted elements	0
Remesh failed parts independently	Off
Reuse mesh for identical bodies	Off
Number of bodies that have reused mesh	0
Time to complete mesh(hh:mm:ss)	00:00:17
Computer name	

Appendix 2 – Temperature model for thermal simulation

Temperature Model for Lunar Lander
Solar Flux at 1 AU = 1361.166 W/m^2
Earth Albedo = 0.340
Shape = Sphere
Cross sectional Area = 1.000 m^2
Material Emissivity = 0.924
Material Absorptivity = 0.248
Internal Power Dissipation = 0.000 W

Appendix 3 – Thermal analysis equations for boiloff calculations

Eq. 1 below is the heat equation for change of state and can be used to calculate boiloff rates for propellant tanks of a given mass. Where Q is the energy required to vaporize a substance, m is the mass of the propellant, and L is the latent heat of vaporization. The time derivative of this equation gives the rate of heat input required to cause this phase change.

$$Q = mL \; [1]$$

Appendix 4 – SolidWorks Mass and Size Properties

Mass properties of NewAssembly-Titanium
Configuration: Default
Coordinate system: -- default --

Mass = 23796.55 grams

Volume = 5432263.08 cubic millimeters

Surface area = 538795.82 square millimeters

Center of mass: (millimeters)
X = 1506.51
Y = 1698.63
Z = 2801.30

Principal axes of inertia and principal moments of inertia: (grams * square millimeters)
Taken at the center of mass.

Ix = (0.00, 0.92, 0.40)	Px = 96003092.82
Iy = (0.00, -0.40, 0.92)	Py = 677880202.46
Iz = (1.00, 0.00, 0.00)	Pz = 735734754.42

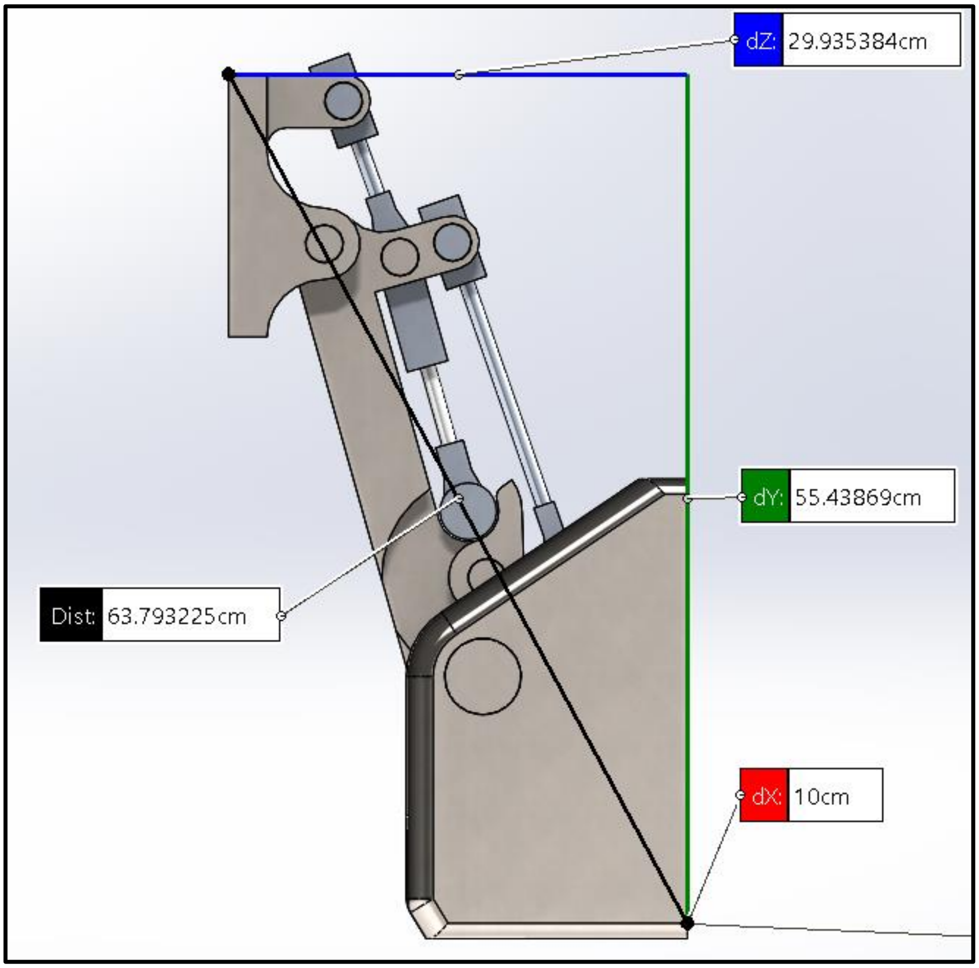
Moments of inertia: (grams * square millimeters)
Taken at the center of mass and aligned with the output coordinate system. (Using positive tensor notation.)

Lxx = 735734754.42	Lxy = 5.26	Lxz = 13.73
Lyx = 5.26	Lyy = 188757707.24	Lyz = 212998517.47
Lzx = 13.73	Lzy = 212998517.47	Lzz = 585125588.05

Moments of inertia: (grams * square millimeters)
Taken at the output coordinate system. (Using positive tensor notation.)

Ixx = 256135740620.88	Ixy = 60895443706.93	Ixz = 100425803164.39
Iyx = 60895443706.93	Iyy = 240935131874.47	Iyz = 113446120731.81

Appendix 5 – SolidWorks Model with Size Properties



Appendix 6 – SolidWorks Thermal Simulation Settings

Fuel Tank Material	7050-T73510 Aluminum Alloy
MAST Material	Ti-5Al-2.5Sn Titanium Alloy
Fuel Tank Initial Temperature	-253 °C
Latch Initial Temperature	-210 °C
MAST Initial Temperature	-73 °C
Incoming Heat Flux	1,410 W/m^2