LAUNCH VEHICLE MGC-A1

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EGR 101 SECTION - 15

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Introduction

Objectives:

The development of a new launch vehicle is a significant undertaking that requires careful consideration of several critical factors. One such factor is the need to design a vehicle capable of carrying a specific size and weight of payload into Low Earth Orbit (LEO). In this project, there is a strong emphasis on simplicity and cost-effectiveness to ensure the vehicle is accessible to a broad range of users. Achieving this goal will require researching and applying engineering solutions that streamline the launch process and minimize expenses without compromising the safe and efficient delivery of the payload. With these objectives in mind, the development of a new launch vehicle is a challenging endeavor that requires an innovative mindset.

Requirements:

The vehicle must be designed with two to three stages to meet the specified requirements. Sequential staging must be the only method used in the vehicle, which is a process of jettisoning expended sections of the rocket during flight to optimize the efficiency of the remaining stages. The exclusive use of solid propellants is also mandatory in the vehicle. These propellants are commonly used in the aerospace industry due to their high energy output and ease of storage, which makes them ideal for low-cost propulsion. Following these guidelines, the vehicle will be capable of efficient, reliable, and safe space travel.

Constraints:

The design of a launch vehicle must consider the specific dimensions and weight of the payload to ensure a successful launch. For the MGC-A1, the rocket must be able to accommodate a cylindrical payload with a diameter of 6 feet and a height of 7 feet. Additionally, the rocket must be capable of delivering payloads weighing between 600-700lbs to an orbit height of 100-200nmi (LEO).

Propulsion System Design

Introduction to launch vehicle:

The MGC-A1 is a three-stage rocket designed to launch a small reconnaissance satellite into Low Earth Orbit (LEO). The rocket can accommodate a payload with a minimum weight of 600 lbs. and a maximum weight of 700 lbs. To house the payload, the rocket's payload bay must accommodate a cylinder of 6 feet in diameter and a height of 7 feet. The orbit altitude must be between 100-200 nmi, and the rocket is set to launch from Kennedy Space Center (KSC), Florida. Additionally, the rocket's free velocity has been specified as 1,341(ft/sec).

Payload selection:

To ensure the successful launch of a payload, it is necessary to have a launch vehicle optimized for the specific weight and dimensions of the payload. The MGC-A1 rocket is a launch vehicle designed to carry a payload mass of 650 lbs. The payload mass was chosen due to it being in the middle of the range given by the client which allows for a 50 lb tolerance from the minimum and maximum mass. This also allows for structural weight to be added in form of thickness to structural components protecting the payload and its journey to Low Earth Orbit. The payload bay is specifically designed to accommodate a cylinder with a diameter of 6 feet and a height of 7 feet. These parameters must be met to ensure the safe and efficient transportation of the payload into orbit.

Orbit selection and velocity required:

Achieving the desired orbit altitude requires careful calculation and precise engineering of the launch vehicle. For the MGC-A1, the target orbit altitude is 150 nmi, which has been factored into the rocket's design. This orbit altitude was chosen because 150 nmi is directly in between the range of 100 nmi to 200 nmi specified by the client. Having the target orbit altitude in the middle of the required range ensures it is not to low or to high for Low Earth Orbit which allows a slight margin of error in case the actual orbital velocity deviates from the target orbit velocity. This must be accounted for to ensure that Low Earth Orbit is achieved, but also not over-shot resulting in the payload deployed being in an incorrect altitude. Additionally, the required velocity for the rocket to reach this altitude has been calculated to be 25,385.07853 ft/s. These values have been considered during the development of the MGC-A1 to ensure the rocket can reach the desired orbit altitude with the necessary velocity.

Orbital Velocity Calculation:

$$V_0 = (R_E)\sqrt{\frac{g_0}{(R_E + h)}\frac{ft}{s}}$$

$$V_0 = (20.902\ \times 10^6\ ft)\sqrt{\frac{32.174\ \frac{ft}{s^2}}{\left(20.902 \times 10^6\ ft\ +\ \left(150\ nmi \times 6076.12\frac{ft}{nmi}\right)\right)}}$$

$$V_0 = 25385.07853\ \frac{ft}{s}$$

V_0 is the orbit velocity, in $\frac{ft}{s}$

R_E is the radius of the Earth =$20.902 \times 10^6 ft$

g_0 is the acceleration due to gravity = $32.174\frac{ft}{s^2}$

h is the orbit altitude above Earth's surface, in ft

nmi represents the unit of nautical mile: $1nmi = 6{,}076.12ft$

Delta V Required

The Kennedy Space Center in Florida is the launch site that will be utilized per request from the client. Additionally, the launch site's proximity to the Atlantic Ocean provides a safe trajectory for the rocket's ascent. Free velocity refers to the velocity that a spacecraft would have if it were moving solely under the influence of gravity. It is the velocity that would allow an object to escape the gravitational pull of a planet or other celestial body without any additional propulsion. In the context of the MGC-A1, the free velocity associated with the launch site is 1,341 ft/s. The theoretical delta V calculated is 29606.83736 ft/s.

Delta V – Theoretical Free Space:

$$\Delta V = \sqrt{1.25\left(V_{orbit}^2 + 2gh - V_i^2\right)\frac{ft}{s}}$$

$$\Delta V = \sqrt{1.25\left(\left(25385.07853\ \frac{ft}{s}\right)^2 + 2\left(32.174\ \frac{ft}{s^2}\right)\left(150\ nmi \times 6076.12\ \frac{ft}{nmi}\right) - \left(1341\ \frac{ft}{s}\right)^2\right)\frac{ft}{s}}$$

$$\Delta V = 29606.83736\ \frac{ft}{s}$$

V_{orbit} is the orbital velocity calculated above, in $\frac{ft}{s}$

g is the acceleration due to gravity = $32.174\frac{ft}{s^2}$

h is the orbital altitude in ft

V_i is the free velocity associated with the launch site

Launch Vehicles Researched and Reasoning for Initial Estimates:

The design of a launch vehicle requires a deep understanding of the requirements for the successful delivery of payloads to their intended destination. To optimize the design of the launch vehicle, research was conducted on existing vehicles that share similar characteristics with the desired specifications. These vehicles were selected based on criteria such as total weight, number of stages, and payload weight, all of which are critical factors in the design of an efficient and effective launch vehicle. Through careful analysis of these existing vehicles, the aim was to gain insights into the best practices and design elements that could be applied to the MGC-A1.

Launch Vehicles Researched:

Name	Total Weight	# Of Stages	Payload [LEO]	MOTOR #1	MOTOR #2	MOTOR #3	MOTOR #4
Electron (USA/NZ)	28,660 lbs.	2	660 lbs.	Rutherford X 9 [Liquid]	Rutherford X 1 [Liquid]	N/A	N/A
Long March I (CHIN)	179,830 lbs.	3	660 lbs.	YF-2A [Liquid]	YF-3A [Liquid]	FG-02 [Solid]	N/A
Small Satellite Launch Vehicle (IND)	260,000 lbs.	4	1,100 lbs.	S85 [Solid]	S7 [Solid]	S4 [Solid]	16 X 50N Bipropellant Thruster [Maneuvering]
Vegas (EUR)	302,000 lbs.	4	≈3,150 lbs.	P80 [Solid]	Zefiro 23 [Solid]	Zefiro 9 [Solid]	AVUM [Liquid]
Delta K (Just for Motor reference for ISP)	N/A	N/A	N/A	AJ10-118K (Aerojet N2O4) Solid	N/A	N/A	N/A

Preliminary Research Conclusion:

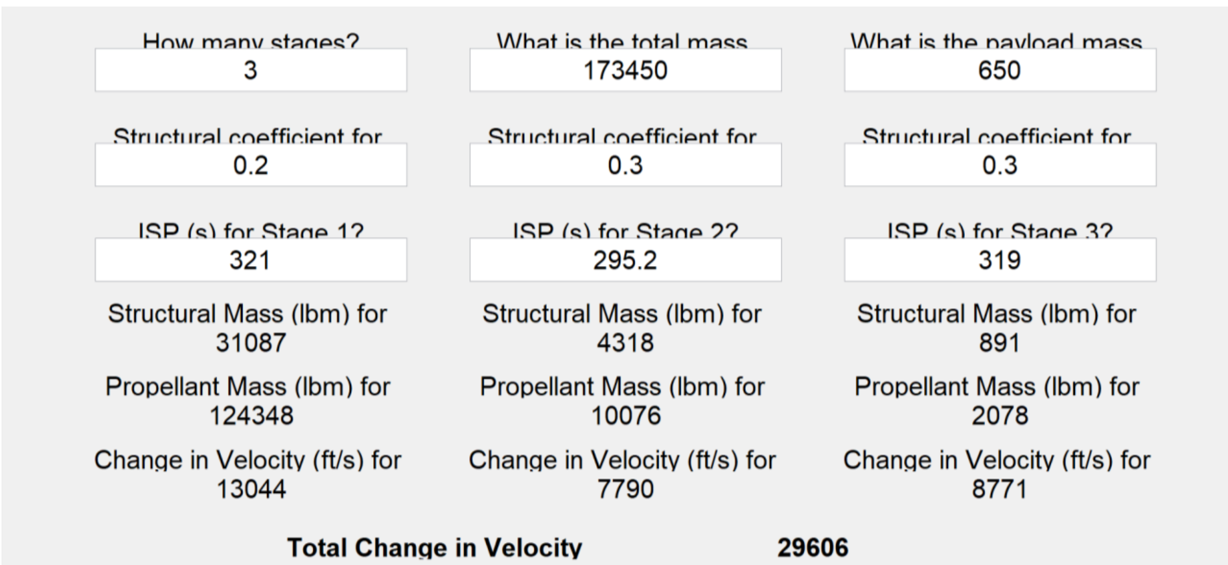
During the research on existing rockets, analysis was done on a range of vehicles that shared similar characteristics to the desired specifications. Through this analysis, it was found that the Vegas Rocket provided useful information for the design. This rocket utilizes the Zefiro 23 motor, which is the same motor that was selected for the first stage. Additionally, the Vegas Rocket has a similar payload weight to the MGC-A1, which drew conclusions on the payload bay design and weight distribution. Although the Long March Rocket has a similar number of stages and payload weight, it utilizes all liquid motors, which is not aligned with the requirements for solid propellant use. The Electron rocket has a comparable payload to the desired specifications. However, it differs from the design in that it only has two stages and uses liquid motors instead of solid propellants. The Delta K rocket provided valuable insights into the specific impulse (ISP) of certain motors, while the Small Satellite Launch vehicle gave useful information on solid motors. By analyzing the Delta K rocket, information was collected on the efficiency of various motors and how they could be utilized in the design. Similarly, the Small Satellite Launch vehicle showed the advantages and challenges associated with the use of solid propellants, which are key components of the launch vehicle design. Overall, the analysis of these rockets provided valuable information that helped to inform design decisions and optimize the launch vehicle for success.

MATLAB input output (first estimate):

Introduction:

A MATLAB program is utilized, which optimizes the weight distributions among the selected number of stages. It is emphasized that the MATLAB program optimizes the stage mass fractions assuming that the stages are fired and jettisoned in sequence and not in parallel. This optimizer will not inform whether the inputs are perfect for the rocket to work. However, it can tell what the best possible achievable Delta V would be from the inputs. The program optimizes mass fractions for each stage using the following inputs: Desired number of stages (2 or 3), Planned specific impulse for each stage (seconds), Total vehicle mass, including propellant, structural, and payload mass (lbm), Payload mass (lbm), excluding propellant or structural mass, Structural coefficients for each stage.

First trial of using the MATLAB optimizer:



Conclusion

During the first trial of the MATLAB optimizer, the rocket was configured with three stages after research showed that a rocket with this configuration would be more efficient and reliable. The total mass of the rocket was determined to be 173450 lbs. with the payload mass set at 650 lbs. as previously determined. The structural coefficients for each stage were estimated through trial and error due to a lack of precise values, while the ISP values were obtained from existing rocket motors. Although the values used in the calculations were based on theoretical assumptions, the resulting percentage difference in Delta V was found to be zero, indicating that the rocket would perform as expected. However, it should be noted that achieving a motor with a structural coefficient of 0.3 and an ISP (specific impulse) of over 300 for certain stages was not feasible in practical terms. Upon further analysis, it was also discovered that the stage three motor did not meet the necessary requirements for a third stage motor.

MATLAB input/output (Final Estimate):

Introduction:

The final estimate generated by the MATLAB optimizer was obtained through a process of brainstorming and iteration. This process involved adjusting various parameters that impacted the rocket's components to optimize its overall performance. In particular, the total weight of the rocket and the structural coefficients, as these were crucial factors that could significantly affect the rocket's performance. Careful consideration and adjustment of the parameters, particularly the total weight of the rocket and the structural coefficients, resulted in a design that met the desired specifications with an impressive accuracy. Specifically, the difference in Delta V was found to be only 0.01%.

Second and final trial of the MATLAB optimizer:

How many stages?	What is the total mass (lbm)?	What is the payload mass (lbm)?
3	101000	650
Structural coefficient for Stage 1?	Structural coefficient for Stage 2?	Structural coefficient for Stage 3?
0.2	0.2	0.2
ISP (s) for Stage 1?	ISP (s) for Stage 2?	ISP (s) for Stage 3?
289	297	287
Structural Mass (lbm) for Stage	Structural Mass (lbm) for Stage	Structural Mass (lbm) for Stage
16398	3119	552
Propellant Mass (lbm) for Stage	Propellant Mass (lbm) for Stage	Propellant Mass (lbm) for Stage
65594	12477	2209
Change in Velocity (ft/s) for	Change in Velocity (ft/s) for	Change in Velocity (ft/s) for
9755	10217	9638
Total Change in Velocity (ft/s):	**29609**	

Conclusion:

The chosen values were thought of in considering the weight of the rocket and how many stages there would be. The MGC-A1 is just 101,000 lbs. in weight and achieves a Delta V of 29609 ft/s, which is only 0.01% off from the calculated Delta V of 29606 ft/s

Trajectory spreadsheet:

Introduction:

The trajectory spreadsheet is designed by Embry-Riddle Aeronautical University professors to enable calculations of finding the thrust needed for each stage and provides other information like the flight path and change in angle of the rocket during flight through graphs. (Refer to Appendix A) The Trajectory utilizes the calculations done by MATLAB for the weight allotments and therefore it makes initial design ideas such as frontal area, thrust to weight ratio, pitch over maneuver altitude and initial inclination angle. The initial coefficients of drag were estimated by researching existing launch vehicles and by using the coefficient of drag graph. (Refer to Appendix E) The initial frontal area was estimated using the diameter of the payload as the largest diameter for the cross-plane analysis. Reasonable T/W ratios were determined using data from existing launch vehicles as well as the motor chart. (Refer to Appendix B)

Thrust to weight:

Stages	T/W used (lbf/lbm)
Zefiro 23 (1st stage)	2.6
M56A-1 (2nd stage)	2.96
Pegasus 3 (3rd stage)	2.5

Conclusion:

To achieve a successful rocket launch, precise adjustments to the thrust to weight ratios to meet the target values for final thrust and burn times for each stage must be met. They must ensure that the actual capabilities of the motor are within 10% of the target values. These adjustments generate trajectory graphs that provide a visual representation of the flight path, altitude, and other critical parameters of the launch vehicle, including its final velocity, angle, and amount of drag. Additionally, adjusting the rocket's angle after each stage to reach the specified altitude while optimizing the smoothness of the theta vs. time graph and trajectory graph must be done. Careful planning and execution of these adjustments are crucial to optimizing the rocket's performance and increasing the likelihood of mission success. In summary, designing and launching a successful rocket requires meticulous attention to critical parameters and precise adjustments to optimize its trajectory. In this trajectory spreadsheet, most of the data was inputted through pattern recognition, in which each input had a different effect on some form of output. Therefore, manipulating most of the inputs brought the best results that were needed for the success of MGC-A1.

Motor selection:

Introduction:

The thrust, ISP, burn time and structural mass input into the MATLAB and Trajectory Spreadsheet were based off current existing motors. (Refer to Appendix B) With these values input into the trajectory spreadsheet, three solid motors were found to be most ideal for this launch vehicle.

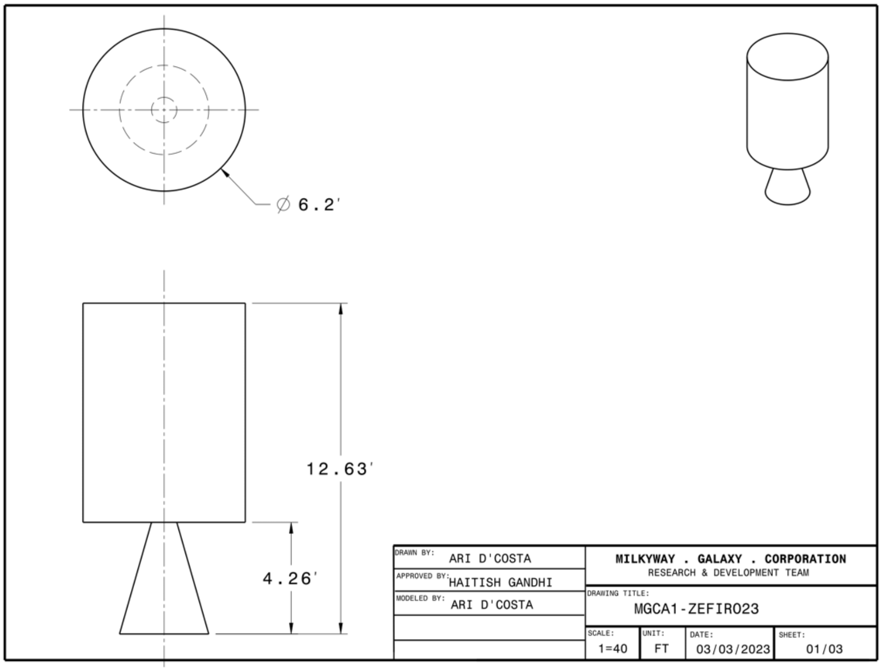
Motor selected and values according to the tables:

Motors	Thrust	ISP	Burn time	Height (ft)	Diameter (ft)	Total weight	Propellant weight
Zefiro 23 (Stage 1)	269,700	289	72	12.63	6.2	59,300	52,700
M56A-1 (Stage 2)	51,369	297	60	12.99	3.7	11,390	10,363
Pegasus 3 (Stage 3)	7,778	287	68	6.82	3.18	1,929	1,700

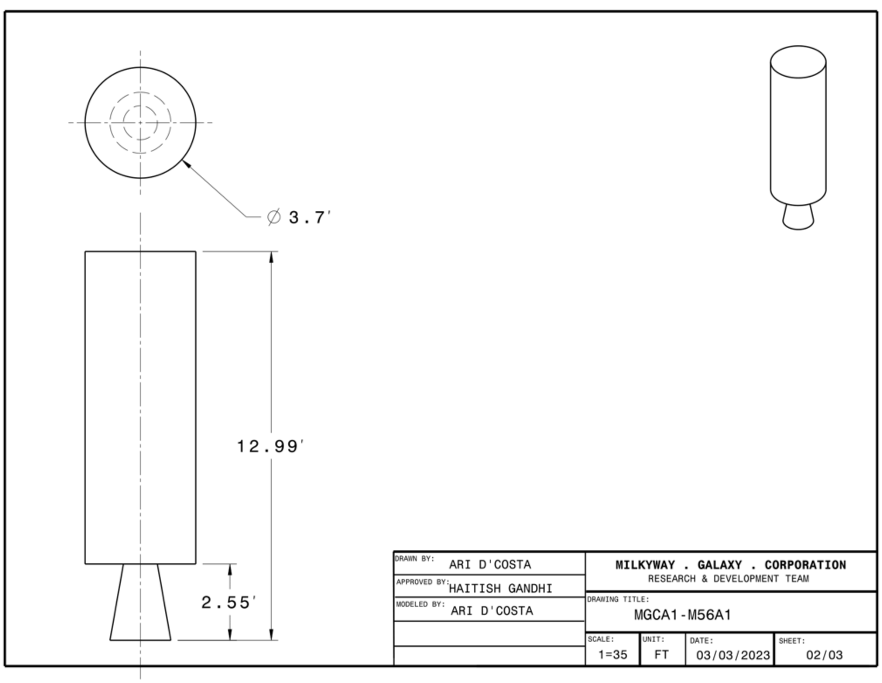
Motor values according to trajectory spreadsheet:

Motors	Thrust	ISP	Burn time	Height (ft)	Diameter (ft)	Total weight	Propellant weight
Zefiro 23 (Stage 1)	262,600	289	72.2	12.63	6.2	59,300	52,700
M56A-1 (Stage 2)	56,264	297	65.9	12.99	3.7	11,390	10,363
Pegasus 3 (Stage 3)	8,530	287	74.3	6.82	3.18	1,929	1,700

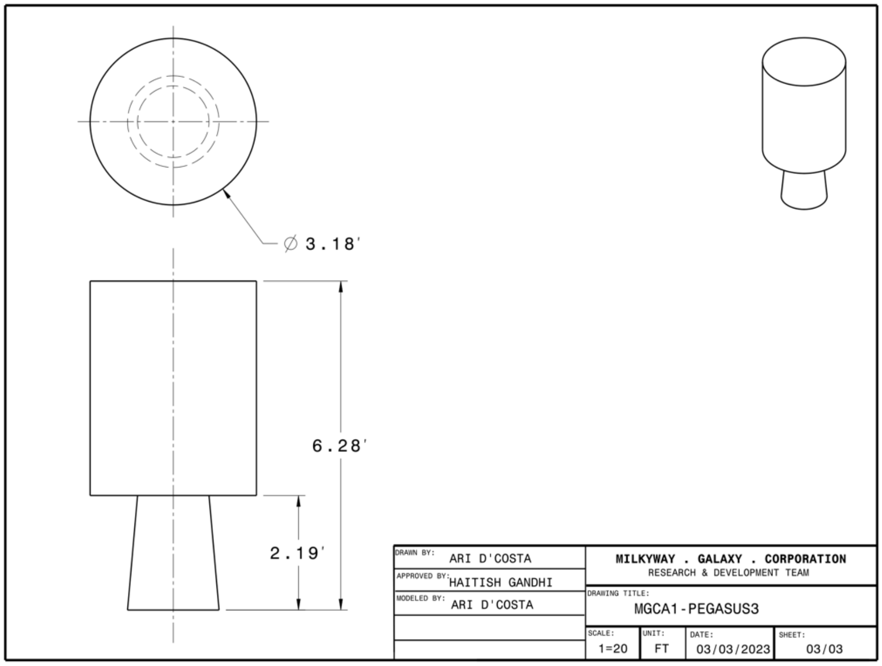
ZEFIRO 23 MOTOR DRAFTING



M56-A1 MOTOR DRAFTING



PEGASUS 3 MOTOR DRAFTING



Conclusion:

The MGC-A1 launch vehicle will feature a multi-stage configuration, with each stage designed to utilize a specific motor to optimize performance. The Zefiro 23 motor is the ideal choice for the first stage due to its high reliability and proven track record in successful launches. The M56A-1 motor will be employed in our second stage to provide additional power and thrust needed to reach the desired altitude. Finally, the Pegasus 3 motor is selected for the third stage, which will propel the payload into its final orbit. Each motor has been selected based on its specific performance characteristics, with careful consideration given to weight, thrust, and cost. By combining these motors into a multi-stage configuration, the MGC-A1 will be an efficient and cost-effective launch system capable of delivering the specified payload into LEO.

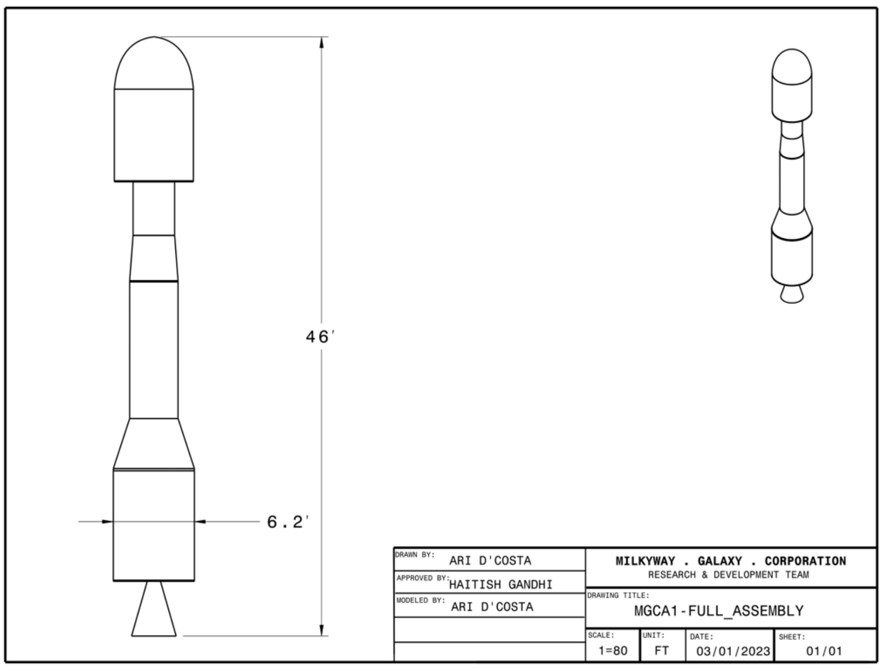
Structural System design

Introduction:

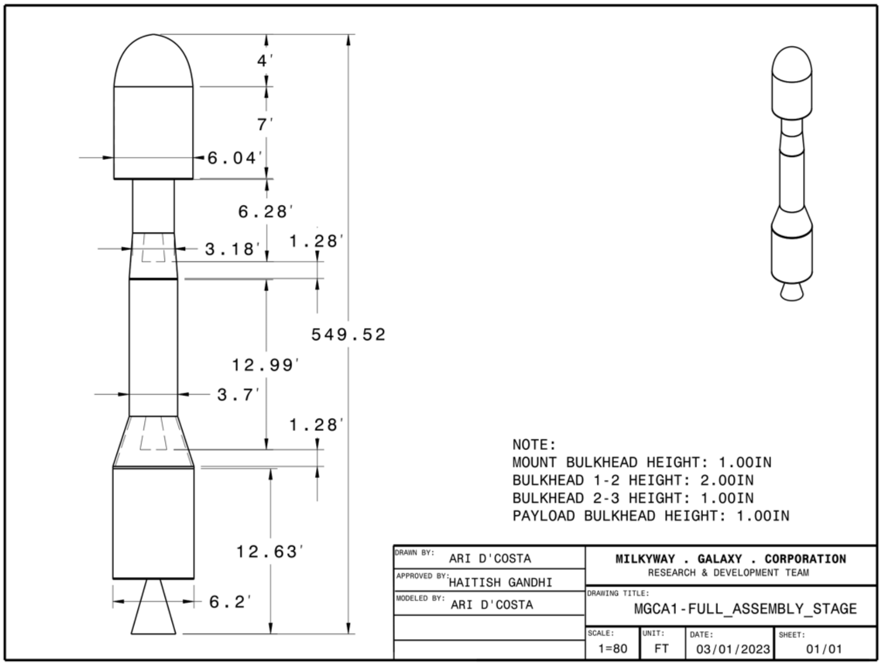
The MGC-A1 uses several structural components within its design. The three main structures are the fairings, the bulkheads, and the nosecone. These structures ensure the safe transportation of the payload by protecting the payload from the high stress and heat of the exterior environment, minimizing stress on internal structures, and supporting the overall weight as well as frame.

Catia Models for Assembly:

FULL ASSEMBLY DRAFTING



FULL ASSEMBLY WITH STAGING DRAFTING



Fairing Calculations:

Introduction:

The fairings help to make the Launch Vehicle aerodynamic, reducing the drag and improving overall stability. Additionally, the fairings encapsulate other internal parts like the motor nozzles and payload to shield them against aerodynamic stresses and high external heat. There are four total fairings on the MGC-A1 as it is a three-stage rocket. There are two interstage fairings which are the interstage fairing from stage one to stage two and the interstage fairing from stage two to stage three. There is also a payload fairing which protects the payload and must fit around the payload given by the client. Lastly, there is a nosecone fairing which is an elliptical shape for the MGC-A1.

Fairings:

FAIRING	THICKNESS (in)	MATERIAL	YIELD STRENGTH (lb/in²)	DIAMETER UPPER (in)	DIAMETER LOWER (in)	HEIGHT (in)
INTERSTAGE 1-2	2	Steel (ASTM A414 GRADE H)	51,000	44.40	74.40	45.89
INTERSTAGE 2-3	1	Steel (ASTM A414 GRADE H)	51,000	38.16	44.40	41.60
PAYLOAD FAIRING	¼	Aluminum (6063-T5)	20,000	72.50	72.50	84
NOSE CONE	¼	Aluminum (6063-T5)	20,000	N/A	72.50	48

Load calculations:

$$LOAD_{fairing} = T\ (lb)$$

$$T = 269{,}700\ lb$$

$$LOAD_{fairing} = 269{,}700\ lb$$

$LOAD_{fairing}$ is the compression load acting on the vehicle in lb

T is the first stage thrust in lb

Minimum Thickness calculations:

$$t_{fairing} = \frac{D_V - \sqrt{D_V^2 - \frac{4 \times LOAD_{fairing}}{\pi \times \sigma_{yield}}}}{2} (in)$$

$$t_{fairing} = \frac{(74.4in) - \sqrt{(74.4\ in)^2 - \frac{4 \times (269{,}700\ lbf)}{\pi \times \left(51{,}000 \frac{lbf}{in^2}\right)}}}{2} (in)$$

$$t_{fairing} = 0.02263\ in$$

D_V is the diameter of the launch vehicle (in)

σ_{yield} is the yield strength (or stress) of the selected material ($\frac{lb}{in^2}$)

$LOAD_{fairing}$ is the compression load calculated (lb)

Volume and weight:

FAIRING NAME	THICKNESS (in)	VOLUME (in^3)	DENSITY (lb/ft^3)	WEIGHT (lbs.)
NOSECONE	¼	1,769.9029	170	174.1223721
PAYLOAD	¼	4,766.5815	170	468.9315214
INTERSTAGE (2-3)	1	5,263.5707	490	1492.74866
INTERSTAGE (1-2)	2	16,548.6472	490	4693

Interstage Fairing Calculation:

$$V = \left(\frac{1}{3}\pi(r_1^2 + r_1 r_2 + r_2^2)h\right) - \left(\frac{1}{3}\pi(r_3^2 + r_3 r_4 + r_4^2)h\right)$$

$$V = \left(\frac{1}{3}\pi((37.2\ in)^2 + (37.2\ in)(22.2\ in) + (22.2\ in)^2)(45.888\ in)\right) - \left(\frac{1}{3}\pi((35.2\ in)^2 + (35.2\ in)(20.2\ in) + (20.2\ in)^2)(45.888\ in)\right)$$

$$V = 129{,}857.8687\ in^3$$

r_1 is the lower radius of the exterior of the fairing in in

r_2 is the upper radius of the exterior of the fairing in in

r_3 is the lower radius of the interior of the fairing in in

r_4 is the upper radius of the interior of the fairing in in

h Is the height of the interstage fairing in in

Nose Cone Volume Calculation:

$$V = \frac{\pi(d_1^2 h_1 - d_1^2 h_2)}{6}$$

$$V = \frac{\pi\left((18in)^2(48in) - (77.75in)^2(47.75in)\right)}{6}$$

$$V = 1769.9029in^3$$

d_1 is the outer diameter at the base of the nosecone in in

d_2 is the inner diameter at the base of the nosecone in in

$h_1$is the outer height of the nosecone in in

h_2Is the inner height of the nosecone in in

Volume of Payload Fairing:

$$V = \pi(r_1^2 - r_2^2)h$$

$$V = \pi((36.25\ in)^2 - (36.00\ in)^2)(84\ in)$$

$$V = 4766.5815\ in^3$$

r_1 is the outer radius of the fairing in in

r_2 is the inner radius of the fairing in in

h Is the height of the fairing in in

Fairing Mass Calculation:

$$Mass\ =\ \rho \times V$$

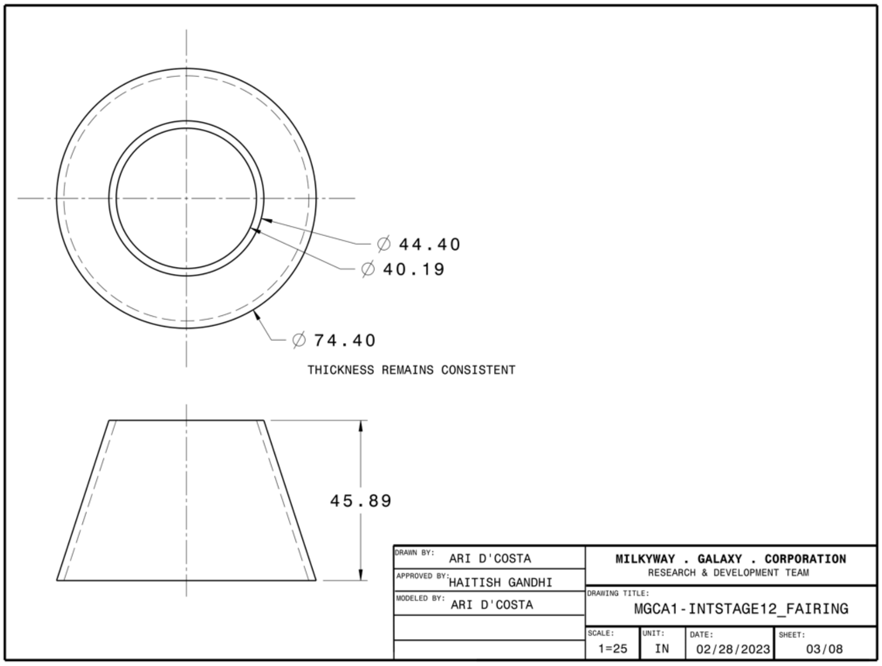
$$Mass = \left(490\frac{lbm}{ft^3}\right) \times \left(16{,}548.6472\ in^3 \times \left(\frac{1\ ft}{12\ in}\right)^3\right)$$

$$Mass = 4{,}693.196\ lbm$$

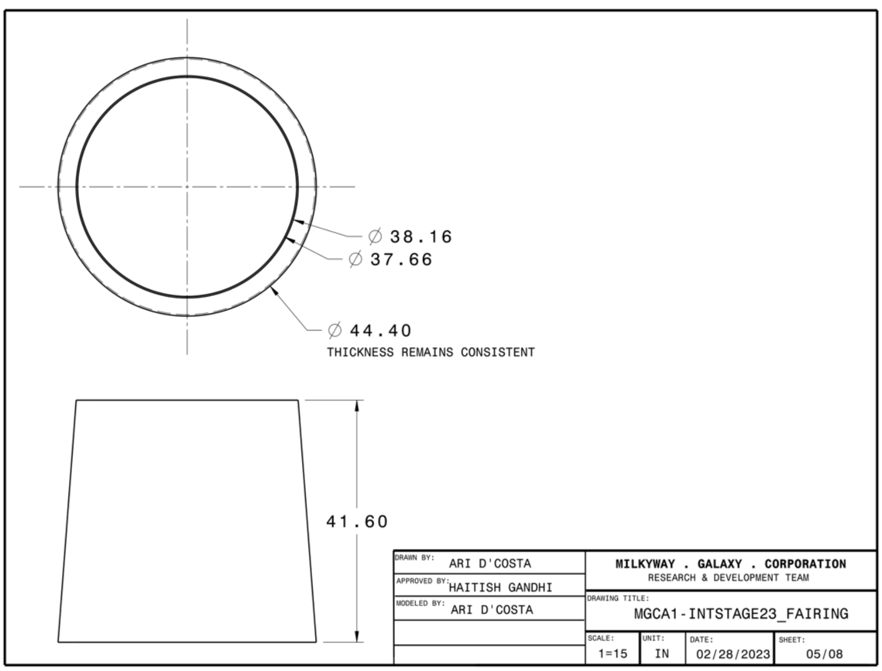
ρ is the density of material in $\frac{lbm}{ft^3}$

V is the volume of the fairing in in^3

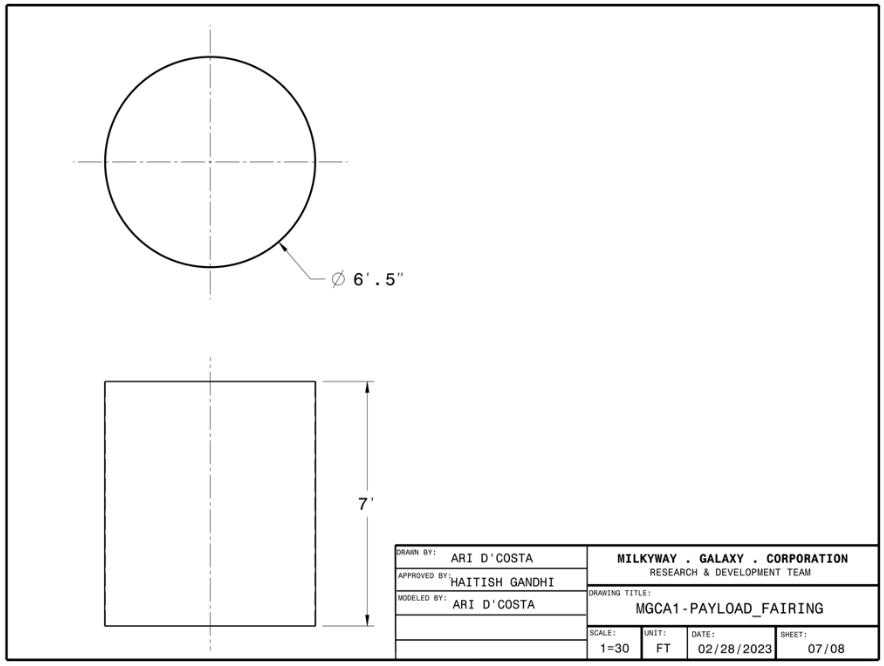
INTERSTAGE 1-2 FAIRING DRAFTING



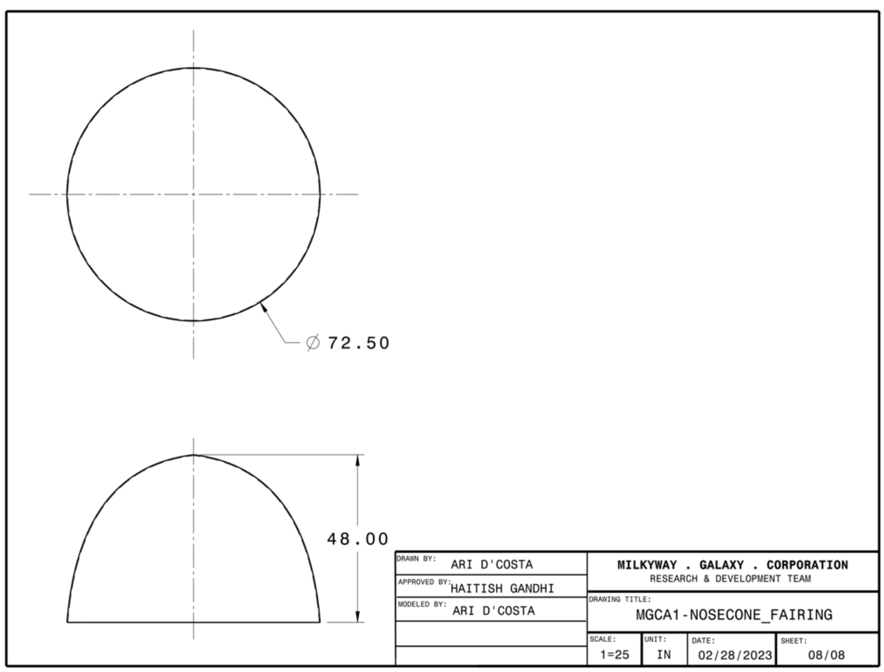
INTERSTAGE 2-3 FAIRING DRAFTING



PAYLOAD FAIRING DRAFTING



NOSE CONE FAIRING DRAFTING



Conclusion:

The MGC-A1 featured four fairings situated in interstage 1-2, interstage 2-3, around the payload, as well as the elliptical nosecone itself. The calculated minimum thickness of the fairings are 0.02 inches however, the client required at least a ¼ in. thickness for all fairings, however structural modifications were made to the MGC-A1 in that interstage fairing 1-2 has a thickness of 2 in. for its wall. Thickness was also added to interstage fairing 2-3, however 1 in. was added to its wall instead due to it supporting less weight and handling less compressive stress. These thickness modifications were made to make the rocket structure more structurally sound, safer, and more reliable in that it can handle higher stress environments ensuring that the launch vehicle will safely transport the valuable payload. For both interstage fairings, a conical cylindrical shape for the frame was used as shown in the CATIA models. All other fairings including the nosecone fairing and payload fairing had a ¼ in. thickness utilized different frame shapes. The nosecone fairing is elliptical for the following reasons: It minimizes air resistance, which improves aerodynamic efficiency, in which the rocket will go faster and use less energy; And it reduces shock waves in which it makes it more stable for high-speed applications. Finally, the cylindrical payload fairing is located around the payload, and it is an essential structure to ensure that the valuable payload is protected from exterior elements including high heat and stress. The elliptical nosecone and payload fairing utilize aluminum while the interstage fairings utilize steel for their structure.

Vehicle bulkheads:

Introduction:

Bulkheads are crucial in rockets because they provide structural support and separation between different components, particularly in multi-stage rockets. They are essentially walls that divide the rocket into compartments, with each compartment serving a specific function. The bulkheads help distribute the weight of the rocket, particularly the upper stages and prevent the payload and other components from collapsing under the rocket's acceleration and vibration during launch. Overall, bulkheads play an essential role in ensuring the safe and efficient operation of rockets, which is crucial for the success of space missions. To achieve this, the MGC-A1 bulkheads will be installed between the launch mount, interstage 1-2, interstage 2-3, and below the payload. The client's requirements dictate that the bulkheads must have a thickness of at least 1 inch or thicker, considering safety and reliability.

Bulkhead Stress and Material Factors

Bulkhead	Load (lb)	T/W at Burnout (lbf/lbm)	Material	Poisson's Ratio (lbf)	Yield Stress (lb/in^2)	Bulkhead Radius (in)	Thickness (in)	Applied Stress (lb/in^2)
Payload	1293.0538	0.3985	Aluminum (6063-T5)	0.330	20,000	36.25	1	318.7438
Interstage 2-3	3628.1895	2.1246	Steel (ASTM A414, Grade H)	0.290	51,000	22.2	1	4793.0354
Interstage 1-2	16949.9816	5.5839	Steel (ASTM A414, Grade H)	0.290	51,000	37.2	2	14965.0973
Mounting	84641.4317	1	Steel (ASTM A414, Grade H)	0.290	51,000	37.2	1	50707.4150

Volume of Mounting Bulkhead Calculation Table with all volumes and weights:

BULKHEAD NAME	VOLUME (ft^3)	DENSITY (lb/ft^3)	WEIGHT (lbs.)
MOUNTING	2.516	490	1232.787
INTERSTAGE 1-2	5.03	490	2465.573
INTERSTAGE 2-3	0.896	490	439.043
PAYLOAD	2.389	170	406.130

Load of Payload Bulkhead:

$$LOAD_{bulkhead} = Total\ Weight\ Above\ Bulkhead\ (lbm)$$

$$LOAD_{bulkhead} = 650\ lbm + 468.9315\ lbm + 174.1224\ lbm$$

$$LOAD_{bulkhead} = 1293.0538\ lbm$$

Volume Bulkhead Calculation:

$$V = \pi r^2 h$$

$$V = \pi \left(37.2in \times \frac{1\ ft}{12\ in}\right)^2 \left(1in \times \frac{1\ ft}{12\ in}\right)$$

$$V = 2.516\ ft^3$$

V is the volume of the bulkhead in ft^3

r is the radius of the bulkhead in in

h is the height of the bulkhead in

Weight of Mounting Bulkhead:

$$Mass\ =\ \rho \times V$$

$$Mass\ =\ \left(490\frac{lbm}{ft^3}\right) \times (2.5159 ft^3)$$

$$Mass\ =\ 1232.7861\ lbm$$

ρ is the density of material in $\frac{lbm}{ft^3}$

V is the volume of the bulkhead in ft^3

Thrust to Weight at Motor Burnout Interstage 2-3:

$$\frac{T}{W}_{at\ motor\ burnout} = \frac{Thrust\ of\ the\ Stage}{Vehicle\ Weight\ at\ Motor\ Burnout}$$

$$\frac{T}{W}_{at\ motor\ burnout} = \frac{51{,}369\ lbf}{24178.4266\ lbm}$$

$$\frac{T}{W}_{at\ motor\ burnout} =\ 2.1246$$

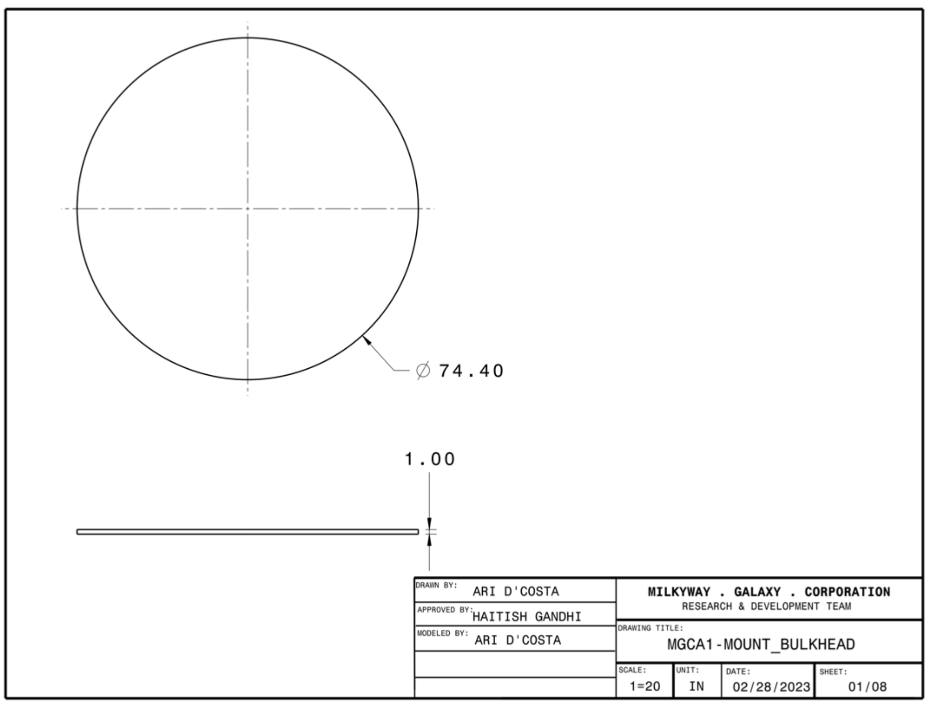
Applied Stress on Interstage 1-2:

$$\sigma_{applied} = \frac{12(LOAD_{bulkhead})\left(\frac{T}{W}\right)_{engineburnout}}{4\pi t_{bulk}^2}\left[(1+v)\ln\left(\frac{a}{\sqrt{0.4a^2+t_{bulk}^2}-0.675t_{bulk}}\right)\right]\left(\frac{lb}{in^2}\right)$$

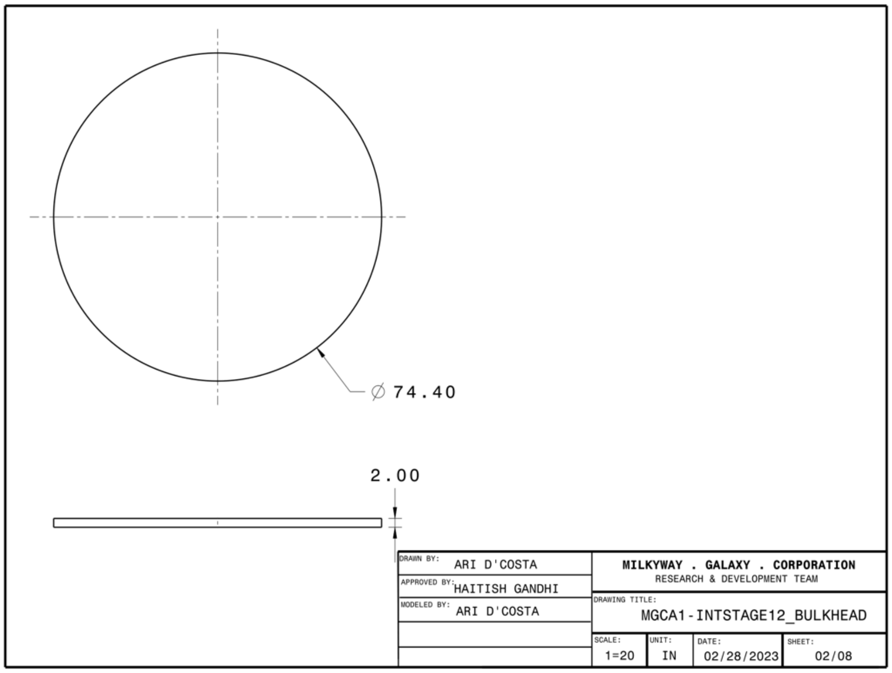
$$\sigma_{applied} = \frac{12(16949.0973lbm)\left(5.5839\frac{lbf}{lbm}\right)}{4\pi(1in)^2}\left[(+0.290lbf)\ln\left(\frac{37.2in}{\sqrt{0.4(37.2in)^2+(2in)^2}-0.675(2in)}\right)\right]\frac{lb}{in^2}$$

$$\sigma_{applied} = 14965.09725\frac{lb}{in^2}$$

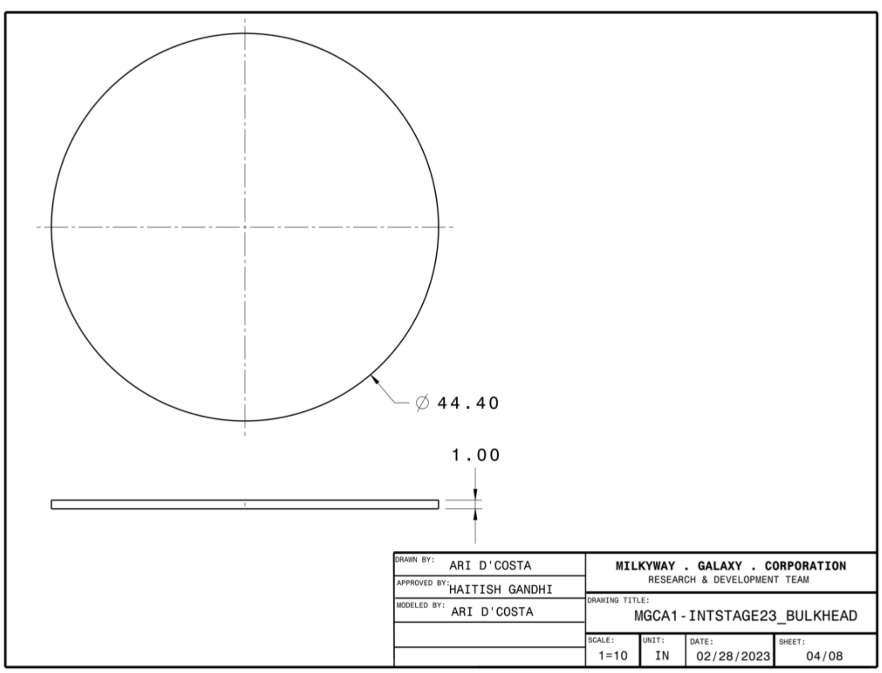
MOUNTING BULKHEAD DRAFTING



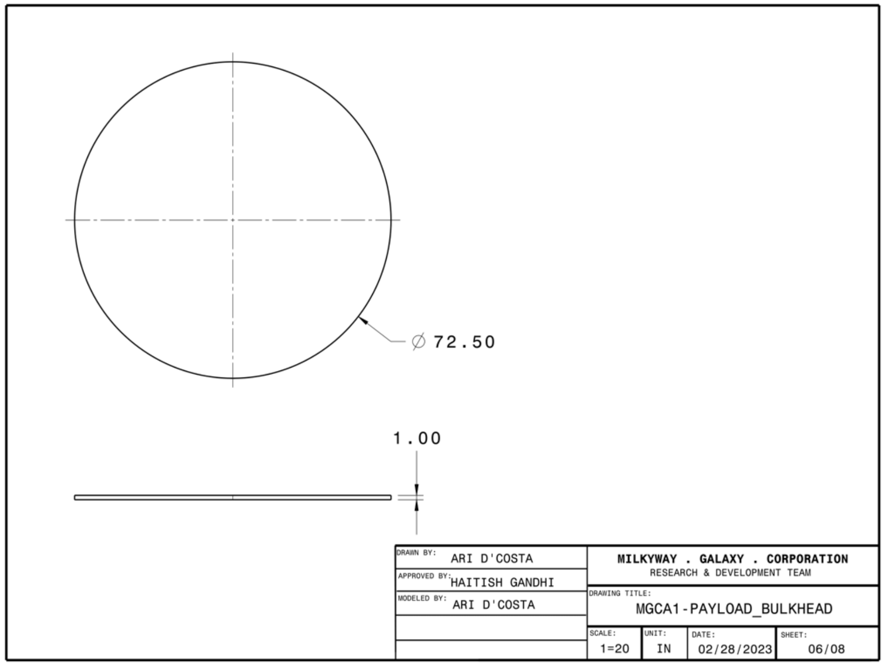
INTERSTAGE 1-2 BULKHEAD DRAFTING



INTERSTAGE 2-3 BULKHEAD DRAFTING



PAYLOAD BULKHEAD DRAFTING



Conclusion:

The MGC-A1 featured four bulkheads situated between the launch mount, interstage 1-2, interstage 2-3, and beneath the payload. While the client mandated a 1-inch thickness for all bulkheads, the interstage 1-2 bulkhead was increased to 2 inches so it could handle higher yield stress which was required because of a higher thrust-to-weight ratio at burnout. This increased interstage 1-2's strength and improved its ability to support the structure above. The payload bulkhead was constructed using aluminum, while the other bulkheads used steel as the primary material.

Material:

Introduction:

The materials utilized to make the structural components of a Launch Vehicle are imperative to the strength of the structure as well as the weight. This directly correlates to how much stress the rockets structures can handle, but also how much it will cost due to the weight added as the density and thickness of the material needed increases. The materials used in the MGC-A1 included Steel and Aluminum with their specifications in the table below.

PROPERTIES	STEEL (ASTM A414, GRADE H)	ALUMINUM (6063-T5)
Density (lb/ft^3)	490	170
Price ($/lb)	57.00	69.00
Yield Strength (lb/in^3)	51,000	20,000

Description:

For the main portion of the MGC-A1 vehicle encompassing the Mounting Bulkhead, Interstage 1-2 Bulkhead and Fairing, and Interstage 2-3 Bulkhead and Fairing, it uses Steel specifically ASTM A414, Grade H. This material proved to be one of the most affordable options while researching materials. Although heavier, it is very strong as shown by its high yield strength and gives the rocket a stronger base structure increasing its safety and reliability while transferring the payload. For all the structural components above the Interstage Fairing 2-3 including the Payload Bulkhead, Payload Fairing, and the Elliptical Nosecone, the MGC-A1 uses Aluminum 6063-T5. This material is still within a lower cost although higher than steel, and has relatively good yield strength, but the main reason it was chosen was because it is lightweight and prevents the rocket from being top-heavy.

Conclusion:

Materials used for the Launch Vehicle has a major effect on the cost of the rocket due to weight added by the material, but also the overall strength of the structure. It is important to note that the MGC-A1 does not sacrifice the strength of the structure for the cost. This was clearly done to maximize the safety and reliability of the rocket allowing it to handle extreme stresses that competitors launch vehicles cannot handle.

Structural Weight:

Introduction:

The structural weight of a rocket is a critical aspect to ensuring the successful launch of the vehicle. The structural weight refers to the components of the rocket that connect and protect the motors and payload. This includes the fairings, bulkheads, and nosecone. Ensuring that enough structural weight is allocated to the correct sections is crucial to the safety and stability of the rocket. The MATLAB stage optimizer has given values that are then input into the trajectory spreadsheet where the trajectory spreadsheet will create a flight path for the rocket.

Weight Per Stage:

NAME	ALLOTED MATLAB (lbs.)	ALLOTTED TRAJECTORY (lbs.)	PERCENTAGE
STAGE #1	16,398	16,000	2.43%
STAGE #2	3,119	3,000	3.82%
STAGE #3	552	1,070	93.8%

Percent Difference Stage 1:

$$Percent\ Difference = \left|\frac{Theoretical\ -\ Actual}{Theoretical}\right| \times 100$$

$$Percent\ Difference = \left|\frac{16398\ lb\ -\ 16000\ lb}{16398\ lb}\right| \times 100$$

$$Percent\ Difference = 2.43\%$$

Conclusion:

The structural allotted weight from MATLAB and the allotted weight for the Trajectory Spreadsheet should be within 20% of each other. (Refer to Appendix A) The structural weight for stage 1 and stage 2 are within the 20% range. The structural weight for stage 3 is not within the 20% range. To address this issue, a decision was made to deviate from the originally calculated values in MATLAB and shift weight from Stage 1 and Stage 2 to Stage 3. A total of 398lb from Stage 1 and 119lb from Stage # were transferred to Stage 3, resulting in a combined weight of 1,069 lb for Stage 3. This value deviates significantly from the MATLAB value, however, it is closer to the actual weight calculated thus making the trajectory spreadsheet more accurate. It should be noted that the allotted weight value input to the trajectory spreadsheet is 1070 lbs. compared to the calculated value of 1069 lb.

Calculate the cross-area:

Introduction:

The cross area or frontal area is the cross-sectional area of the vehicle perpendicular to the velocity vector. The coefficient of drag is the measure of resistance an object encounters as it moves through a fluid. These values are important because they determine how well an object moves through a fluid.

Stage Configurations:

	Diameter (ft)	Vehicle Frontal Area (ft^2)	Coefficient of Drag	Vehicle Fontal Area Trajectory Spreadsheet (ft^2)	Coefficient of drag Trajectory Spreadsheet
Stage 1	6.2	30.2	0.2	30.2	0.2
Stage 2	6.05	28.7	0.2	28.7	0.2
Stage 3	6.05	28.7	0.2	28.7	0.2

Vehicle Frontal Area Stage 1:

$$A = \pi r^2$$

$$A = \pi(3.1ft)^2$$

$$A = 30.2\ ft^2$$

r Is the radius of the largest stage in ft

Conclusion:

The calculated frontal areas of the vehicle were incorporated into the trajectory spreadsheet, resulting in a more precise representation of the launch vehicle's trajectory. It should be noted, however, that the coefficients of drag were predetermined and were not modified in the trajectory spreadsheet. (Refer to Appendix E)

Vehicle Height Determination (Buffer Heights Determined):

Introduction:

The buffer height is a designated height added to the actual height of the launch vehicle. This height added provides a margin of safety to mitigate risks.

Height and Diameter table:

Name	Height (ft)
Zefiro 23	12.63ft
M56A-1	12.99ft
Pegasus 3	6.82ft
Interstage Buffer Height (2x)	1.28ft
Payload	7ft
Nose Cone	4ft
Total	46.00ft

Buffer Height Calculation:

$$H_B = \frac{H_V - (H_M + H_{Pay} + H_{NC})}{H_V}$$

$$H_B = \frac{46\ ft - (32.44 ft + 7\ ft + 4\ ft)}{46\ ft}$$

$$H_B = 5.5652\%$$

H_B is the percentage of the launch vehicle that is the buffer height

H_V is the total height of the launch vehicle in *ft*

H_M is the combined height of all the motors in *ft*

H_{Pay} is the height of the payload in *ft*

H_{NC} is the height of the nose cone in *ft*

Conclusion:

The height of the MGC-A1 was set to 46 ft to ensure that it met the customer's buffer height requirement. This height allowed for a total buffer height percentage of 5.56% which fell within the acceptable tolerance level. Furthermore, this buffer height provided sufficient clearance between the bottom of the motor nozzles and the bulkhead below them.

Conclusion:

The structural design of the MGC-A1 is reliant on the fairings and bulkheads. This includes the weight of them, the materials they are made of the cross-sectional frontal area they encompass which affects drag, and the buffers needed between motors. The bulkheads support the weight of the stages above them, while the fairings provide an efficient aerodynamic shape between stages and shield the payload and motor nozzles from exterior elements. The aerodynamic shape considers the cross-sectional frontal area of the widest part of the Launch Vehicle and the resulting drag which helps to determine the efficiency of the MGC-A1 in the Trajectory Spreadsheet. The shape and aerodynamic efficiency is important, the materials used and strength of those materials are essential. With the use of steel for the main portion of the rocket in combination with increased thickness of fairings, it ensures that the launch vehicle is structurally sound and reliable. Additionally, the use of aluminum for the Elliptical Nosecone and Payload Fairing which also has a high yield strength, ensures the payload is safe and protected during flight. To further ensure that the Launch Vehicles Structure is reliable and safe, a buffer height is added in the interstage to ensure the motor nozzles do not get damaged during flight. All these factors make the MGC-A1 reliable, robust, efficient, and, most importantly, safe for the transportation of the valuable payload.

Integration with Launch Mount

Launch Mount Introduction:

The MGC-A1 will launch from a mount provided by the client. (Refer to Appendix C) The client specified that there are three set positions for the launch mount interface. To ensure safe departure, the MGC-A1 must fit properly onto the provided launch mount or modifications must be made by the client to accommodate the launch vehicle. The MGC-A1 will use the 30-degree position of the launch mount. To use the launch mount three checks must be done: vehicle diameter check, nozzle diameter check, and nozzle height check.

Integrations: Calculations

Stage 1:

	Motor Diameter (ft)	Nozzle Diameter (in)	Nozzle Height (in)	Mounting Bulkhead Diameter (in)
Zerifo 23	6.2	40.92	51.15	74.4

Launch Mount:

	T-bracket	T-bracket Distance (Center to Center)	T-bracket Distance (Inner to Inner)	T-bracket Distance (Outer to Outer)	Launch Pad Height
Launch Mount	12.375 in	54 in	41.625 in	66.375 in	92.25 in

Nozzle Diameter Calculation:

$$Diameter = 0.55 \times D$$

$$Diameter = 0.55 \times (6.2ft) \times \left(\frac{12\ in}{1\ ft}\right)$$

$$Diameter = 40.92\ in$$

D is the motor diameter in ft

Nozzle Height Calculation:

$$Height\ = 1.25 \times D_{Nozzle}$$

$$Height\ = 1.25 \times (40.92\ in)$$

$$Height\ = 51.15\ in$$

D_{Nozzle} is the diameter of the nozzle in ft

Inner distance between two opposing T-brackets Calculation:

$$D_{Inner} = D_{Center} - L_{Tbracket}$$

$$D_{Inner} = (54\ in) - (12.375\ in)$$

$$D_{Inner} = 41.625\ in$$

D_{Center} is the distance between the center of two opposing T-brackets in in

$L_{Tbracket}$is the length of one T-bracket in in

Outer distance between two opposing T-brackets Calculation:

$$D_{Outer} = D_{Center} + L_{Tbracket}$$

$$D_{Outer} = (54\ in) + (12.375\ in)$$

$$D_{Outer} = 66.375\ in$$

D_{Center} is the distance between the center of two opposing T-brackets in in

$L_{Tbracket}$ is the length of one T-bracket in in

Launch Mount Checks:

Vehicle diameter check: Mounting Bulkhead Diameter > T-bracket distance (Outer to Outer)

: 74.4in > 66.375in

Nozzle diameter check: T-bracket distance (Inner to Inner) > Nozzle diameter

: 41.625in > 40.92in

Nozzle height check: Launch pad height > Nozzle height.

: 92.25in > 51.15in

Conclusion:

The MGC-A1 launch vehicle will utilize the first configuration of the launch pad, where the T-brackets are set to be 30 degrees from the vertical. The vehicle diameter check concluded that the launch vehicle would comfortably fit onto the launch pad. The nozzle diameter check concluded that the nozzle for stage one will fit between the T-brackets, however there would be less than half an inch clearance on either side of the nozzle relative to the T-brackets. The nozzle height check concluded that the nozzle height is comfortably within the limits of the launch pad configuration.

Systems Integration

Introduction:

The Systems integration section refers to the parameters and tolerances for specific values in the design of the launch vehicles design. (Refer to Appendix D) These values must be met to ensure the safe transport and delivery of the specified payload.

Required Theoretical Free Delta Velocity and Achievable Delta V:

Introduction:

The calculated theoretical delta V and the value in MATLAB must be within 1% of each other. The calculated theoretical delta V is 29606.83736 ft/s. The delta V achieved is 29609 ft/s. Our calculated percent difference is 0.0073%, which is within the tolerance range of 1%.

Calculation for Tolerance:

$$Percent\ Difference = \left|\frac{Theoretical\ -\ Actual}{Theoretical}\right| \times 100$$

$$Percent\ Difference = \left|\frac{29606.83736\ \frac{ft}{s} - 29609\ \frac{ft}{s}}{29606.83736\ \frac{ft}{s}}\right| \times 100$$

$$Percent\ \ Difference\ =\ 0.0073\%$$

Orbit Velocity and Final Velocity:

Introduction:

The orbit velocity and the final velocity must be within 5% of each other. The calculated orbit velocity is 25,385.07853 ft/s. The final velocity achieved is 24124 ft/s. Our calculated percentage difference is 4.97%, which is within the tolerance range of 5%.

Calculation for Tolerance:

$$Percent\ Difference = \left|\frac{Theoretical\ -\ Actual}{Theoretical}\right| \times 100$$

$$Percent\ Difference = \left| \frac{25385.07853\ \frac{ft}{s} - 24124\ \frac{ft}{s}}{25385.07853\ \frac{ft}{s}} \right| \times 100$$

$$Percent\ Difference = 4.97\%$$

Angle at final stage burnout:

Introduction:

The angle at final stage burnout must be between 80 and 90 degrees. The launch vehicle achieved a final angle of 87 degrees which is within the required range of 80 – 90 degrees.

Angle Check:

$$Limit_{lower} < Angle_{final} < Limit_{higher}$$

$$80° < 87.0° < 90°$$

$Limit_{lower}$ is the minimum angle required.

$Angle_{final}$ is the actual final angle of the launch vehicle.

$Limit_{higher}$Is the maximum angle allowed.

Altitude at final stage burnout:

Introduction:

The altitude at the final stage burnout must be between 100–200 nmi. The achieved value is 151.5 nmi. The altitude achieved at final stage burnout is 151.5 nmi. This is within the range of 100-200 nmi.

Altitude Check:

$$Limit_{lower} < Altitude_{final} < Limit_{higher}$$

$$100\ nmi < 151.5\ nmi < 200\ nmi$$

$Limit_{lower}$ is the minimum altitude required.

$Altitude_{final}$ is the actual final altitude of the launch vehicle.

$Limit_{higher}$ is the maximum altitude allowed.

Required thrust of engine(s) and actual thrust of engine(s):

Introduction:

The required thrust of the engines must be within 10% of the actual thrust values of the motors. Our calculated percent difference is 2.53% for stage 1, 9.53% for stage 2, 9.668% for stage 3. All these values are within the tolerance range of 10%.

Motor selected and values according to the tables:

Motors	Thrust (lb)
Zefiro 23 (Stage 1)	269,700
M56A-1 (Stage 2)	51,369
Pegasus 3 (Stage 3)	7,778

Motor values according to trajectory spreadsheet:

Motors	Thrust (lb)
Zefiro 23 (Stage 1)	262,600
M56A-1 (Stage 2)	56,264
Pegasus 3 (Stage 3)	8,530

Tolerances:

Motors	Thrust Percent Difference
Zefiro 23 (Stage 1)	2.53
M56A-1 (Stage 2)	9.53
Pegasus 3 (Stage 3)	9.668

Calculations for Tolerance:

Thrust Calculation: Zefiro 23

$$Percent\ Difference = \left|\frac{Theoretical - Actual}{Theoretical}\right| \times 100$$

$$Percent\ Difference = \left|\frac{269700\ lb - 262{,}600\ lb}{269700\ lb}\right| \times 100$$

$$Percent\ Difference = 2.53\%$$

ISP of selected engine and ISP inputted into MATLAB:

Introduction:

The specific impulse (ISP) value of the selected engine and the ISP input into MATLAB must be within 10% of each other. Our calculated percent difference is 0% for all stages. All these values are within the tolerance range of 10%.

Motor Values According to Tables:

Motors	ISP
Zefiro 23 (Stage 1)	289
M56A-1 (Stage 2)	297
Pegasus 3 (Stage 3)	287

Motor Values According to Trajectory Spreadsheet:

Motors	ISP
Zefiro 23 (Stage 1)	289
M56A-1 (Stage 2)	297
Pegasus 3 (Stage 3)	287

Tolerances:

Motors	ISP %
Zefiro 23 (Stage 1)	0.00
M56A-1 (Stage 2)	0.00
Pegasus 3 (Stage 3)	0.00

Zefiro 23 ISP Tolerance:

$$Percent\ Difference = \left|\frac{Theoretical - Actual}{Theoretical}\right| \times 100$$
$$Percent\ Difference = \left|\frac{289\,sec - 289\,sec}{289\,sec}\right| \times 100$$
$$Percent\ Difference = 0\%$$

Burn time in TRAJECTORY spreadsheet and actual motor burn times:

Introduction:

The burn times input into the trajectory spreadsheet and the actual burn time of the motors must be within 10% of each other. Our calculated percent difference was 0.27% for stage 1, 9.83% for stage 2, and 9.26% for stage 3. All these calculated values are within the tolerance range of 10%.

Motor values according to the tables:

Motors	Burn time (s)
Zefiro 23 (Stage 1)	72
M56A-1 (Stage 2)	60
Pegasus 3 (Stage 3)	68

Motor values according to trajectory spreadsheet:

Motors	Burn time (s)
Zefiro 23 (Stage 1)	72.2
M56A-1 (Stage 2)	65.9
Pegasus 3 (Stage 3)	74.3

Tolerances:

Motors	Burn Time Percent Difference
Zefiro 23 (Stage 1)	0.27 %
M56A-1 (Stage 2)	9.83 %
Pegasus 3 (Stage 3)	9.26 %

Zefiro 23 Burn times Tolerance:

$$Percent\ Difference = \left|\frac{Theoretical\ -\ Actual}{Theoretical}\right| \times 100$$

$$Percent\ \ Difference\ = \left|\frac{72\ sec - 72.2\ sec}{72\ sec}\right| \times 100$$

$$Percent\ \ Difference\ =\ 0.27\%$$

T/W ratios for each stage within allowable range:

Introduction:

The thrust-to-weight ratio for all stages must be between 1 and 3. For the first stage, the thrust-to-weight ratio used was 2.6 which is within the allowable range. For the second stage, the thrust-to-weight ratio used was 2.96, which is within the allowable range. For the third stage, the thrust-to-weight ratio used was 2.5 which is within the allowable range. Although the calculated thrust to weights is different from the thrust to weights used, as long as the thrusts to weights are all within range then the difference is negligible.

Thrust to weight:

Stages	Thrust (lbf)	Weight (lbm)	T/W calculated (lbf/lbm)	T/W used (lbf/lbm)
Zefiro 23 (1st stage)	269,700	101,000	2.67	2.6
M56A-1 (2nd stage)	51,369	19,406	2.64	2.96
Pegasus 3 (3rd stage)	7,778	3,929	1.98	2.5

Thrust to weight calculation for first stage:

$$\frac{T}{W} = \frac{T_{motor}}{W_{stage}}$$

$$\frac{T}{W} = \frac{269{,}700lbf}{101{,}000lbm}$$

$$\frac{T}{W} = 2.67\frac{lbf}{lbm}$$

T_{motor} is the thrust of the motor in lbf

W_{stage} is the weight of the stage lbm

Vehicle H/D ratios:

Introduction:

The launch vehicle meets the required height-to-diameter ratio of 7-15 for the first stage firing configuration with a value of 7.006. The launch vehicle meets the recommended height-to-diameter ratio of 5-13 for the second stage with a value of 5.101. The launch vehicle does not meet the recommended height-to-diameter of 3-11 ratio for the third stage firing configuration with a value of 2.95. Although the height-to-diameter ratio for the third stage firing configuration is not within range, there are negligible effects on the drag coefficient due to the rocket's altitude at the time of the third stage ignition.

Height and Diameter table:

Name	**Height (ft)**
Zefiro 23	12.63ft
M56A-1	12.99ft
Pegasus 3	6.82ft
Interstage Buffer Height (2x)	1.28ft
Payload	7ft
Nose Cone	4ft
Total	46.00ft
Name	**Diameter (ft)**
Zefiro 23	6.2ft
Payload	6.05ft

Final Checks table:

Final Checks	Range	Actual
First Stage Firing Configuration (Entire Vehicle Height)	7 < H/D < 15	7.419
Second Stage Firing Configuration (Stage 2 and everything above it)	5 < H/D < 13	5.101
Third Stage Firing Configuration (Stage 3 and everything above it)	3 < H/D < 11	2.95

Height to Diameter Ratio Calculation:

$$\frac{H}{D} = \frac{Total\ Height}{Diameter}$$

$$\frac{H}{D} = \frac{46.00\ ft}{6.2\ ft}$$

$$\frac{H}{D} = 7.419$$

$Total\ Height$ is the total height of the vehicle

$Diameter$ is the largest diameter of the vehicle

Weight allotments verification:

Introduction:

The allotted weight input in the trajectory spreadsheet and the allotted weight given by MATLAB must be within 20% of each other. The weight allotment for stages 1 and 2 are within the required range. The weight allotment for stage 3 is 131% which is much higher than the allowed tolerance of 20%. The reasoning behind this was that the structural weight for stage 3 was significantly over the allotted weight. To address this issue, a decision was made to deviate from the originally calculated values in MATLAB and shift weight from Stage 1 and Stage 2 to Stage 3. A total of 398lb from Stage 1 and 119lb from Stage # were transferred to Stage 3, resulting in a combined weight of 1,069 lb for Stage 3. This value deviates significantly from the MATLAB value, however, it is closer to the actual weight calculated thus making the trajectory spreadsheet more accurate. It should be noted that the allotted weight value input to the trajectory spreadsheet is 1070 lbs. compared to the calculated value of 1069 lb.

Weight Per Stage:

NAME	ALLOTED MATLAB (lbs.)	ALLOTTED TRAJECTORY (lbs.)	ACTUAL (lbs.)	PERCENTAGE
STAGE #1	16,398	16,000	14,991.35999	8.58%
STAGE #2	3,119	3,000	2958.7920626	5.14%
STAGE #3	552	1,070	1278.1838935	131%

Percent Difference Stage #1:

$$Percent\ Difference = \left|\frac{Theoretical\ -\ Actual}{Theoretical}\right| \times 100$$

$$Percent\ Difference = \left|\frac{16398\ -\ 16000}{16386}\right| \times 100$$

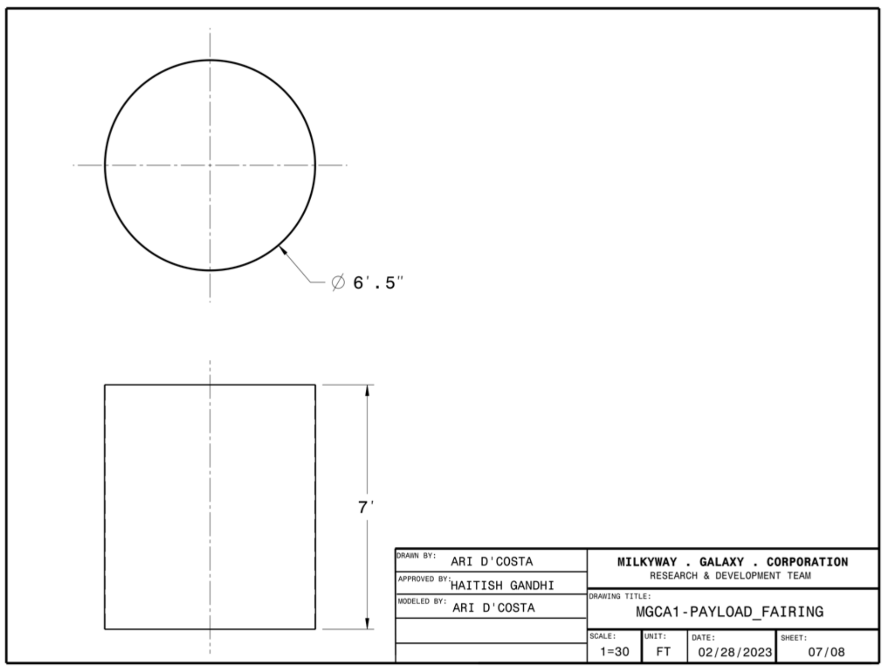
$$Percent\ Difference = 2.49\%$$

Actual payload bay space envelope and payload required space envelope:

Introduction:

The required payload bay space envelope must be able to accommodate a cylindrical payload with a diameter of 6 feet and a height of 7 feet. The payload fairing has an outer radius of 6 feet and 0.5 inches. The thickness of the payload fairing is 0.25 inches. The actual payload bay space envelope is 6 feet in diameter and 7 feet tall. This shows that the payload bay space envelope will only be able to fit the payload if the diameter of the payload is exactly six feet in diameter or less and 7 feet tall or less.

Payload Fairing:



Total Buffer Heights is within the range:

Introduction:

The total of all buffer heights must be no more than 25% or no less than 5% of the total vehicle height. The percentage of the buffer heights is 5.57% of the total vehicle height which is within the 5% - 25% range.

Height Table:

Name	**Height (ft)**
Zefiro 23	12.63ft
M56A-1	12.99ft
Pegasus 3	6.82ft
Interstage Buffer Height (2x)	1.28ft
Payload	7ft
Nose Cone	4ft
Total	46.00ft

Calculations:

$$Percent = \left|\frac{Partial}{Total}\right| \times 100$$

$$Percent = \left|\frac{2 \times 1.28\,ft}{46.00\,ft}\right| \times 100$$

$$Percent = 5.57\%$$

<u>Drag coefficient and frontal area (per stage) in TRAJECTORY spreadsheet is consistent with stage diameters in vehicle fairing calculations:</u>

Introduction:

The Drag coefficient and the frontal area per stage input into the trajectory spreadsheet must be consistent with the stage diameters in the vehicle fairing calculations. All the values input into the trajectory spreadsheet are consistent with the calculated values.

Stage Configurations:

	Diameter (ft)	**Vehicle Frontal Area (ft^2)**	**Coefficient of Drag**	**Vehicle Fontal Area Trajectory Spreadsheet (ft^2)**	**Coefficient of drag Trajectory Spreadsheet**
Stage 1	6.2	30.2	0.2	30.2	0.2
Stage 2	6.05	28.7	0.2	28.7	0.2
Stage 3	6.05	28.7	0.2	28.7	0.2

Frontal Area of Stage 1 Calculation:

$$A = \pi(r)^2$$

$$A = \pi\left(\frac{6.2\,ft}{2}\right)^2$$

$$A = 30.2\,ft^2$$

r Is the radius of the largest stage in ft

Launch Mount Configuration Check:

Introduction:

The launch mount configuration check determines if the given launch mount is suitable to utilize. The vehicle diameter check concluded that the launch vehicle would fit onto the launch pad. The nozzle diameter check concluded that the nozzle for stage one will fit between the T-brackets. The nozzle height check concluded that the nozzle height is within the limits of the launch pad configuration.

Stage 1:

	Motor Diameter (ft)	Nozzle Diameter (in)	Nozzle Height (in)	Mounting Bulkhead Diameter (in)
Zerifo 23	6.2	40.92	51.15	74.4

Launch Mount:

	T-bracket	T-bracket Distance (Center to Center)	T-bracket Distance (Inner to Inner)	T-bracket Distance (Outer to Outer)	Launch Pad Height
Launch Mount	12.375 in	54 in	41.625 in	66.375 in	92.25 in

Nozzle Diameter Calculation:

$$Diameter = 0.55 \times D$$

$$Diameter = 0.55 \times (6.2ft) \times \left(\frac{12\,in}{1\,ft}\right)$$

$$Diameter = 40.92\,in$$

D Is the motor diameter in ft

Nozzle Height Calculation:

$$Height = 1.25 \times D_{Nozzle}$$

$$Height = 1.25 \times (40.92\ in)$$

$$Height = 51.15\ in$$

D_{Nozzle} is the diameter of the nozzle in in

Inner distance between two opposing T-brackets Calculation:

$$D_{Inner} = D_{Center} - L_{Tbracket}$$

$$D_{Inner} = (54\ in) - (12.375\ in)$$

$$D_{Inner} = 41.625\ in$$

D_{Center} is the distance between the center of two opposing T-brackets in in

$L_{Tbracket}$ is the length of one T-bracket in in

Outer distance between two opposing T-brackets:

$$D_{Outer} = D_{Center} + L_{Tbracket}$$

$$D_{Outer} = (54\ in) + (12.375\ in)$$

$$D_{Outer} = 66.375 in$$

D_{Center} is the distance between the center of two opposing T-brackets in in

is $L_{Tbracket} in$

Launch Mount Checks:

Vehicle diameter check: Mounting Bulkhead Diameter > T-bracket distance (Outer to Outer)

: 74.4in > 66.375in

Nozzle diameter check: T-bracket distance (Inner to Inner) > Nozzle diameter

: 41.625in > 40.92in

Nozzle height check: Launch pad height > Nozzle height.

: 92.25in > 51.15in

Cost of Vehicle Accounts for all Components:

Introduction:

The cost of the entire vehicle must account for every individual part of the launch vehicle. To ensure that every structure of the launch vehicle was included in the cost, an itemized receipt has been written concluding that every individual structure has been accounted for.

Analysis:

Component	Cost ($)
Interstage 1-2 Adapter Fairing	$267,512.19
Interstage 1-2 Adapter Fairing	$85,086.67
Payload Protective Fairing	$ 32,356.27
Nose Cone Fairing	$ 12,014.44
Mounting Bulkhead	$ 70,268.84
Interstage 1-2 Bulkhead	$ 140,537.68
Interstage 2-3 Bulkhead	$ 25,025.47
Payload Bulkhead	$ 28,023.36
Structural Manufacturing	$ 7,960,776.53
Solid Motor	$ 4,932,705.00
Propellant	$ 647,630.00
Total Launch Vehicle Cost	$ 14,201,936.47
Cost per LB of Payload	$ 21,849.13

Conclusion:

The launch vehicle meets all required tolerances except for one structural weight allotment tolerance. This structural weight allotment issue pertains to the actual weight of stage 3 being about 1200 lbs. while the allotted structural weight given by MATLAB was 552lbs. To solve this issue, about 500 lbs. of allotted structural weight from stage 1 and 2 were shifted to the allotted structural weight of stage 3. This allowed the allotted structural weight input into the trajectory spreadsheet to be 1070 lbs. which solved the issue of our stage three 3 actual structural weight being about double the MATLAB allotted structural weight.

Cost calculations

Introduction:

As the demand for space travel and exploration continues to grow, the cost of launch services has become a crucial factor in the industry. In order to accurately estimate the cost of providing such services, it is important to consider various factors including the cost of manufacturing and assembly of the vehicle and motor. One common approach to determine this cost is by using the cost of similar existing vehicles and motors to generate a value in dollars per pound. The cost of the empty vehicle, motor thrust, and propellant weight have been determined through this method. However, it is important to note that only the weights of the designed structural components are considered in the structural components cost calculation. This helps estimate the cost that would be charged to the customer in order to provide the launch service that this vehicle could offer. (Refer to Appendix F)

Main calculations:

Structural Cost:

$$Structural\ Cost\ =\ \left(W_{structural}(lb) \times \left(\frac{700}{1lb}\right)\right) + (W_{Material}\ (lb) \times Cost_{material})$$

$$Structural\ Cost\ =\ \left(11{,}372.537(lb) \times \left(\frac{700}{1lb}\right)\right) + (1049.189(lb) \cdot 69)$$

$$+\ (10323.348(lb)\ \cdot\ 57)$$

$$Structural\ Cost\ =\ \$8{,}621{,}601.10$$

$W_{structural}$ is the weight in lb.

$W_{Material}$ is the weight in lb.

Motor Cost:

$$Motor\ Cost\ =\ \left(T_{total}(lb)\right) \cdot \left(\frac{15}{lb}\right)$$

$$Motor\ Cost\ =\ \left(328847(lb)\right) \cdot \left(\frac{15}{lb}\right)$$

$$Motor\ Cost\ =\ \$4{,}932{,}705$$

T_{total} is the total thrust of the motors in lb.

Propellant Cost:

$$Propellant\ Cost = (Propellant_{total}(lb)) \cdot \left(\frac{10}{lb}\right)$$

$$Propellant\ Cost = (52{,}700\ +\ 10{,}363\ +\ 1{,}700) \cdot \left(\frac{10}{lb}\right)$$

$$Propellant\ Cost = \$647{,}630$$

Total Cost:

$$Total\ Cost = Structural\ Cost +\ Motor\ Cost +\ Propellant\ Cost$$

$$Total\ Cost =\ 8{,}621{,}601.1\ +\ 4{,}932{,}705\ + 647{,}630$$

$$Total\ Cost = \$14{,}201{,}936.47$$

Cost Per Pound of Payload:

$$Cost\ per\ Payload = \frac{Total\ Cost}{Payload_{weight}(lb)}$$

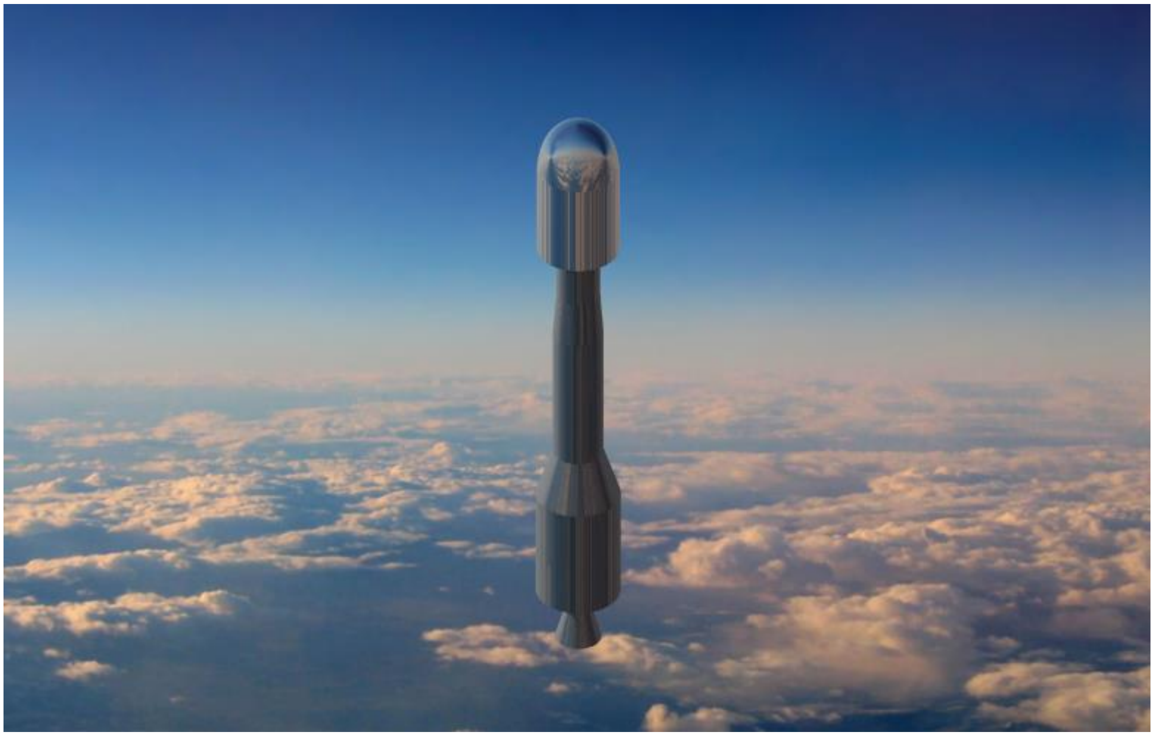
$$Cost\ per\ Payload = \frac{14{,}201{,}936.47}{650}$$

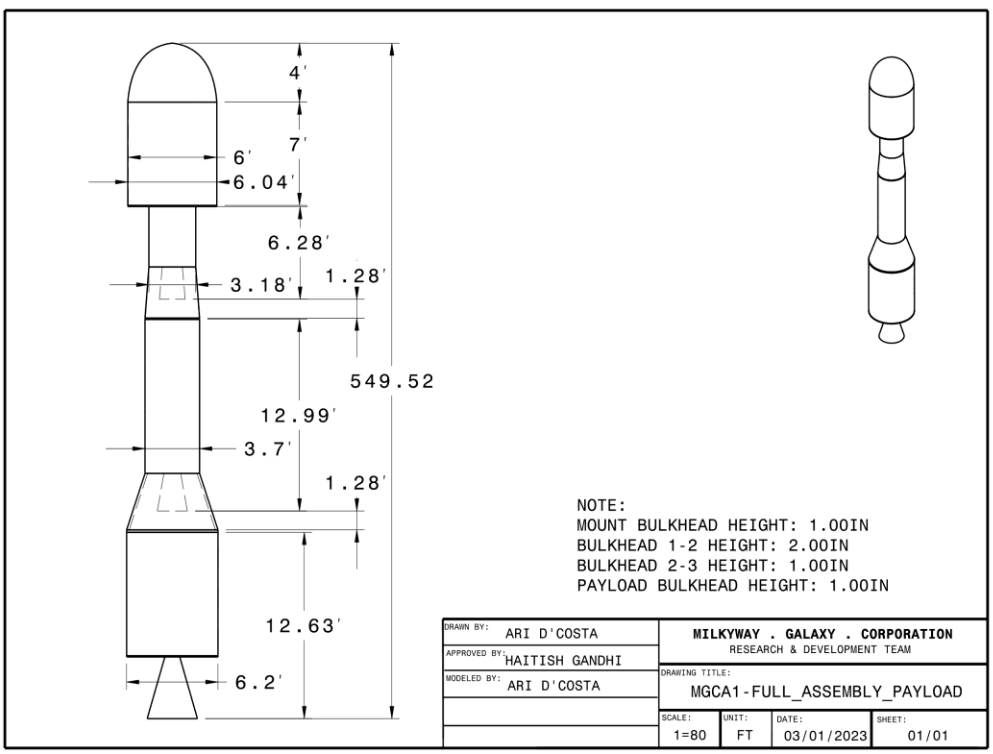
$$Cost\ per\ Payload = \$21{,}849.13$$

Conclusion:

The total cost of the launch vehicle is \$14,201,936.47, which consists of three individual components added together: structural cost, motor cost, and propellant cost. By considering each of these elements, a comprehensive estimate can be made for the customer. Additionally, the cost per pound of payload is \$21,849.13, allowing for a more precise estimate for the customer in terms of the cost required to deliver a specific amount of payload. With this information, the customer can make decisions regarding the associated cost of the launch.

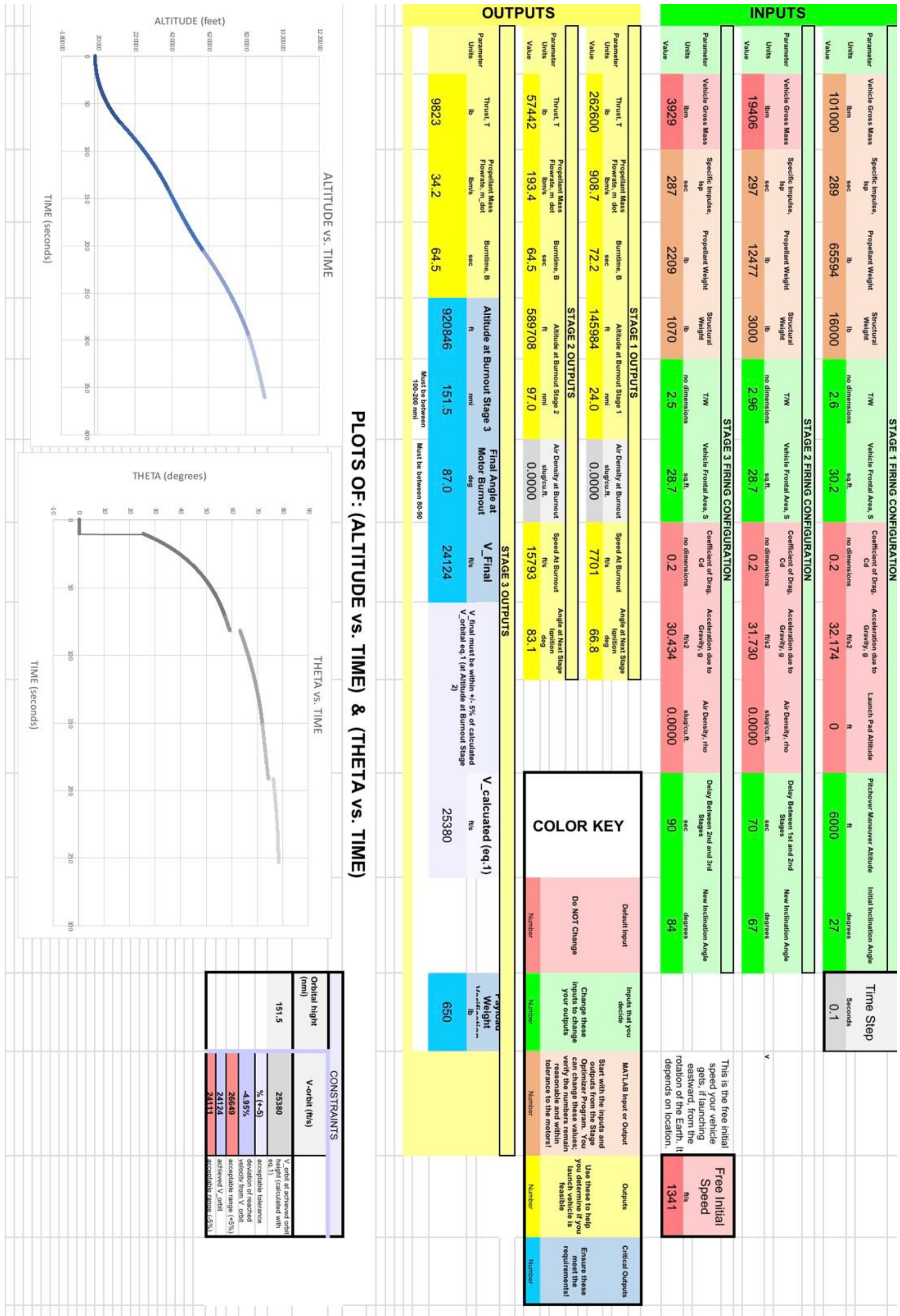
Final drawing of the launch vehicle





Appendix

Appendix A – Trajectory iterations screenshots:



INPUTS

STAGE 1 FIRING CONFIGURATION

Parameter	Vehicle Gross Mass	Specific Impulse, Isp	Propellant Weight	Structural Weight	T/W	Vehicle Frontal Area, S	Coefficient of Drag, Cd	Acceleration due to Gravity, g	Launch Pad Altitude	Pitchover Maneuver Altitude	Initial Inclination Angle	Time Step
Units	lbm	sec	lb	lb	no dimensions	sq.ft.	no dimensions	ft/s2	ft	ft	degrees	Seconds
Value	101000	289	65594	16000	2.6	30.2	0.2	32.174	0	6000	27	0.1

STAGE 2 FIRING CONFIGURATION

Parameter	Vehicle Gross Mass	Specific Impulse, Isp	Propellant Weight	Structural Weight	T/W	Vehicle Frontal Area, S	Coefficient of Drag, Cd	Acceleration due to Gravity, g	Air Density, rho	Delay Between 1st and 2nd Stages	New Inclination Angle
Units	lbm	sec	lb	lb	no dimensions	sq.ft.	no dimensions	ft/s2	slug/cu.ft.	sec	degrees
Value	19406	297	12477	3000	2.96	28.7	0.2	31.730	0.0000	70	67

STAGE 3 FIRING CONFIGURATION

Parameter	Vehicle Gross Mass	Specific Impulse, Isp	Propellant Weight	Structural Weight	T/W	Vehicle Frontal Area, S	Coefficient of Drag, Cd	Acceleration due to Gravity, g	Air Density, rho	Delay Between 2nd and 3rd Stages	New Inclination Angle
Units	lbm	sec	lb	lb	no dimensions	sq.ft.	no dimensions	ft/s2	slug/cu.ft.	sec	degrees
Value	3929	287	2209	1070	2.5	28.7	0.2	30.434	0.0000	90	84

This is the free initial speed your vehicle gets, if launching eastward, from the rotation of the Earth. It depends on location.

Free Initial Speed
ft/s
1341

OUTPUTS

STAGE 1 OUTPUTS

Parameter	Thrust, T	Propellant Mass Flowrate, m_dot	Burntime, B	Altitude at Burnout Stage 1		Air Density at Burnout	Speed At Burnout	Angle at Next Stage Ignition
Units	lb	lbm/s	sec	ft	nmi	slug/cu.ft.	ft/s	deg
Value	262600	908.7	72.2	145984	24.0	0.0000	7701	66.8

STAGE 2 OUTPUTS

Parameter	Thrust, T	Propellant Mass Flowrate, m_dot	Burntime, B	Altitude at Burnout Stage 2		Air Density at Burnout	Speed At Burnout	Angle at Next Stage Ignition
Units	lb	lbm/s	sec	ft	nmi	slug/cu.ft.	ft/s	deg
Value	57442	193.4	64.5	589708	97.0	0.0000	15793	83.1

STAGE 3 OUTPUTS

Parameter	Thrust, T	Propellant Mass Flowrate, m_dot	Burntime, B	Altitude at Burnout Stage 3		Final Angle at Motor Burnout	V_Final		V_calcuated (eq.1)	Payload Weight Verification
Units	lb	lbm/s	sec	ft	nmi	deg	ft/s	V_final must be within +/- 5% of calculated V_orbital eq.1 (at Altitude at Burnout Stage 2)	ft/s	lb
	9823	34.2	64.5	920846	151.5	87.0	24124		25380	650
					Must be between 100-200 nmi	Must be between 80-90				

COLOR KEY

Default Input	Inputs that you decide	MATLAB Input or Output	Outputs	Critical Outputs
Do NOT Change	Change these inputs to change your outputs	Start with the inputs and outputs from the Stage Optimizer Program. You can change these values; verify the numbers remain reasonable and within tolerance to the motors!	Use these to help you determine if you launch vehicle is feasible	Ensure these meet the requirements!
Number	Number	Number	Number	Number

PLOTS OF: (ALTITUDE vs. TIME) & (THETA vs. TIME)

CONSTRAINTS

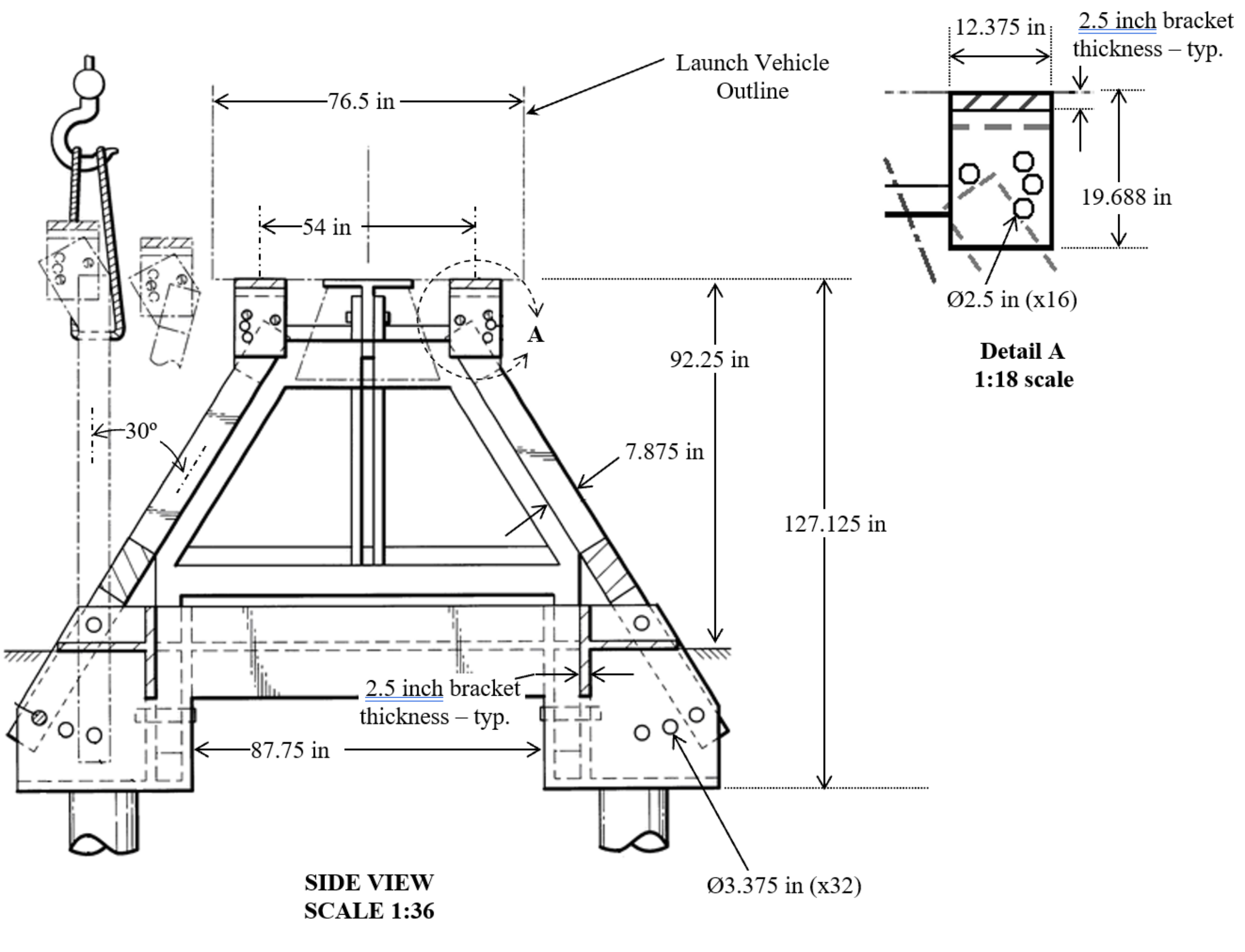
Orbital hight (nmi)	V-orbit (ft/s)	
151.5	25380	V_orbit at achieved orbit height (calculated with eq.1)
	% (+-5)	acceptable tolerance
	-4.95%	deviation of reached velocity from V_orbit
	26649	acceptable range (+5%)
	24124	achieved V_orbit
	24111	acceptable range (-5%)

Appendix B – Motor Charts

Table 1a. Solid Fueled Rocket Engines – Boosters and First Stages							
Thrust (lb)	Engine Designation	Isp (sec)	Designed Burntime (sec)	Height (ft)	Dia. (ft)	Total weight of motor (lb)	Weight of the propellant (lb)
285,500	Aerojet SRB (booster)	275	94	58	5.08	90,000	81,200
155,110	Ariane 3-0 P7.35 (booster)	263	29	27.3	3.51	21,303	16,204
1,455,031	Ariane 5-0 P230 (booster)	286	123	101	9.80	593,000	519,000
370,930	Castor 120 (booster)	280	81	29.59	7.74	117,105	108,097
58,203	Castor 2 (booster)	262	37	19.81	2.59	9,753	8,221
91,542	Castor 4 (first and upper stages)	261	54	29.75	3.34	23,223	20,426
134,840	Castor 4AXL (booster)	269	60	40.29	3.34	32,740	28,942
112,225	GEM 40 (booster)	274	63	42.51	3.34	28,801	25,801
136,706	GEM 46 (booster)	274	75	48.2	3.83	42,520	37,500
191, 425	GEM 60 (boosters)	275	90	53.1	4.98	73,191	66,031
349,938	H-2/J-1-1 (booster)	273	94	59	5.90	156,700	130,500
849,900	M-14 (first stage)	276	46	45.2	8.2	184,210	157,610
178,048	M55/TX-55/Tu-122 (booster)	262	60	24.57	5.47	50,876	45,824
51,369	M56A-1 (first and second stages)	297	60	12.99	3.7	11.390	10,363
683,410	P80 (booster)	280	107	24.6	9.8	209,000	193,600
132,412	Pegasus XL-1 (first stage/air launch)	293	73	29.13	4.16	39,537	33,175
2,589,790	Redesgined SRM (booster)	269	124	126.21	12.62	1,256,398	1,109,998
505,820	SRB-A (booster)	280	101	49.80	8.20	168,400	145,500
269,700	Zefiro 23 (first stage)	289	72	12.63	6.2	59,300	52,700

Table 1b. Solid Fueled Rocket Engines – Upper Stages							
Thrust (lb)	Engine Designation	Isp (sec)	Designed Burntime (sec)	Height (ft)	Dia. (ft)	Total weight of motor (lb)	Weight of the propellant (lb)
2,788	Altair 1 (upper stage)	256	38	6	1.5	524	458
3,102	Altair 1A (upper stage)	255	40	8.2	1.5	850	451
5,500	Altair 3B (upper stage)	289	30	5	1.64	676	604
6,100	Altair 3 (upper stage)	287	28	5	1.64	663	602
17,984	Antares 3 (upper stage)	294	45	10	2.50	3,051	2,835
64,295	Castor 1 (upper stage)	247	27	19.42	2.59	8,492	7,313
91,542	Castor 4 (first and upper stages)	261	54	29.75	3.34	23,223	20,426
117,799	M-23-Mu (upper stage)	285	70	20.3	4.62	28,800	22,700
29,697	M-3B-J (upper stage)	294	87	8.8	4.9	7,900	7,240
279,955	M-24 (upper stage)	288	71	22.3	8.2	75,990	68,480
66,139	M-34 (upper stage)	301	102	11.8	7.2	24,000	21,800
51,369	M56A-1 (first and second stages)	297	60	12.99	3.7	11.390	10,363
10,229	Mage 2 (upper stage)	293	44	4.98	2.52	1,160	1,072
110,224	MIHT-2 (upper stage)	280	64	19.6	5.08	28,000	24,700
55,123	MIHT-3 (upper stage)	280	56	9.8	4.90	13,200	11,000
2,203	MIHT-4 (upper stage)	295	207	8.2	4.5	2,200	1,540
34,508	Pegasus XL-2 (upper stage)	290	73	11.74	4.16	9,548	8,631
7,778	Pegasus-3 (upper stage)	287	68	6.82	3.18	1,929	1,700
4,606	S-30 (upper stage)	275	43	4.1	2.16	831	706
60,181	SR19 (upper stage)	288	66	13.51	4.36	15,502	13,750
70,365	Zefiro 9 (upper stage)	294	117	5.70	6.20	24,400	22,200

Appendix C – Launch Mount Interface



Appendix D – Tolerance Chart

Parameters	**Allowable Range**
Required Theoretical Free ΔV and Achievable Delta V	±1%
V_{orbit} and V_final	±5%
Angle at Burnout	80 to 90 degrees
Final stage Altitude at engine burnout	Within 100 – 200 nmi
I_{sp} of actual selected motor and I_{sp} inputted into TRAJECTORY	±10%
Burntimes of spreadsheet values and actual solid motors	±10%
Required thrust of motor and Actual thrust of motor	±10%
Vehicle H/D ratio (stage 1 firing config., i.e. whole vehicle)	$7 < H/D < 15$

Parameters	**Allowable Ranges**	
Total of all Buffer Heights	no more than 25% of no less than 5% of	Total Vehicle Height
Actual Structural Weight	no more than 20% over at least 80% of	Allotted Structural Weight (MATLAB output)
T/W ratio for any stage	greater than 1.0*	less than or equal to 3.0**

Appendix E – Drag Coefficient Graph

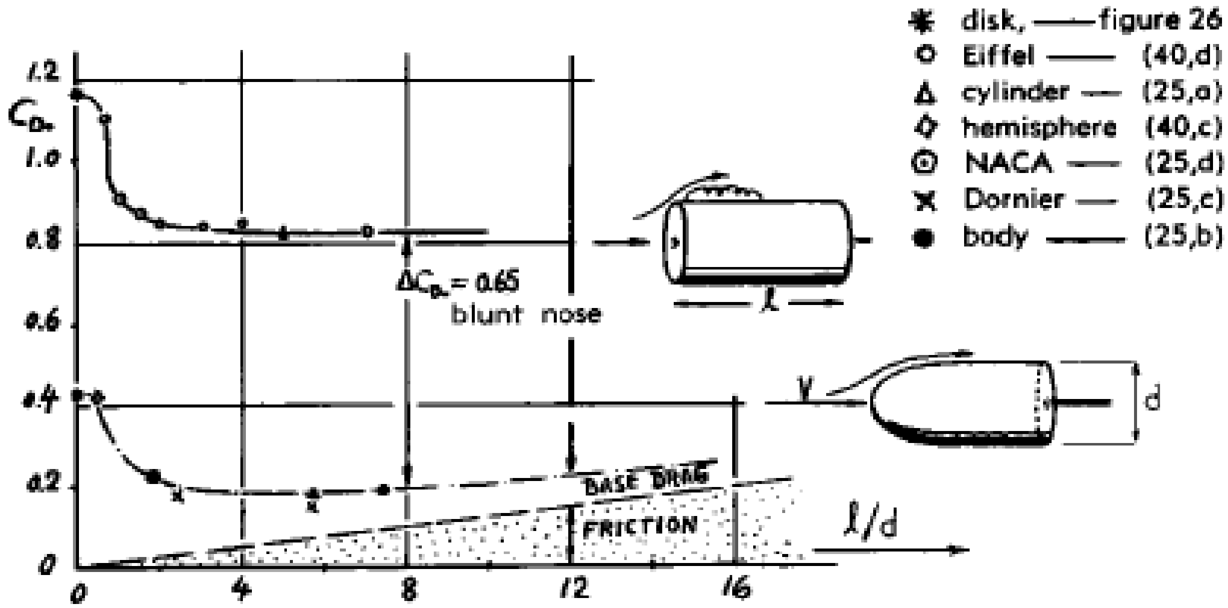


Figure 21. Drag coefficients of cylindrical bodies in axial flow, with blunt shape (in the upper part) and with rounded or streamlined head forms (lower part)—as a function of the fineness ratio l/d.

Appendix F – Preliminary Cost Calculations Excel

Launch Vehicle Preliminary Cost Calculator™ [V1.00]

LICENSED TO: MILKYWAY GALAXY CORPERATION©

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[GENERAL COST CALCULATIONS]

[PAYLOAD]			650.000	Weight (lbs.)

Inter-stage 1-2 Adapter Fairing	$ 57.00	/lb	4693.19635	Weight (lbs.)
Inter-stage 2-3 Adapter Fairing	$ 57.00	/lb	1492.74866	Weight (lbs.)
Payload Protective Fairing	$ 69.00	/lb	468.93152	Weight (lbs.)
Nosecone Fairing	$ 69.00	/lb	174.12237	Weight (lbs.)
Mounting Bulkhead	$ 57.00	/lb	1232.78669	Weight (lbs.)
Interstage 1-2 Bulkhead	$ 57.00	/lb	2465.57330	Weight (lbs.)
Interstage 2-3 Bulkhead	$ 57.00	/lb	439.04340	Weight (lbs.)
Payload Bulkhead	$ 69.00	/lb	406.13561	Weight (lbs.)

Structural Manufacturing ($)	$ 700.00	/lb	11372.5379	Weight (lbs.)
Solid Motor ($)	$ 15.00	/lb	328847.000	Thrust (lbs.)
Propellant ($)	$ 10.00	/lb	64763.000	Weight (lbs.)
Cost Penalty TOTAL ($)	$ -			
Liquid Propellant Engine ($)	$ 20.00	/lb	0	Weight (lbs.)
Liquid Propellant Cost ($)	$ -			

TOTAL LAUNCH VEHICLE COST ($)	$ 14,201,936.47
COST PER LB OF PAYLOAD ($)	$ 21,849.13

[LIQUID MOTOR CALCULATIONS]

	FUEL	OXIDISER		FUEL	OXIDISER	
Motor #1	$ -	$ -	/lb	0.00	0.00	(lbs.)
Motor #2	$ -	$ -	/lb	0.00	0.00	(lbs.)
Motor #3	$ -	$ -	/lb	0.00	0.00	(lbs.)

Motor #1 Total	$ -
Motor #2 Total	$ -
Motor #3 Total	$ -

KEY	INPUT	DEFAULT INPUT	TABLE 2	TABLE 3	MOTOR SCALING	OUTPUTS	CRITICAL OUTPUTS