VENETIA LAUNCH VEHICLE DESIGN

Ad Astra!

Alec Guy Philippe
M.S. Astronautical Engineering
University of Southern California | Viterbi School of Engineering | ASTE-574

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I. Introduction

The design of a launch vehicle (LV) is a complicated process that involves the integration of several engineering fields such as aerospace, materials, and mechanical. This report includes a breakdown of the mathematical design of the Venetia LV. The process begins with identifying the use of the LV. It was determined that the payload would need to be placed in a sun synchronous orbit (SSO). The altitude and inclination of the orbit provides the necessary information to calculate the required ΔV (delta-Velocity) to reach its destination. The next step in the process requires the type of propellant and thruster that would be used. In conjunction with the ΔV requirement, it is now possible to analyze and design the framework of the LV. The final steps include calculating the stresses that the vehicle will encounter, choosing a material that can withstand those stresses, and estimating the cost of the LV program.

For the Venetia LV program, a SSO with an altitude of 741km above the Earth's surface and an inclination of 98.2° was selected. The LV would also be two stage to orbit (TSTO) and would carry a payload of 375kg. Since the selected orbit would be SSO, it can be assumed that the payload would be a satellite for imaging, reconnaissance, or weather predictions. The LV would launch out of the North Sea onboard a large ship. With these requirements, it was then possible to calculate the required orbital parameters, an LV performance summary, and provide a flight profile for the 1st and 2nd steps.

II. Two Stage to Orbit Design

Analysis

With the orbit requirements, it was possible to calculate the required ΔV needed to reach the required orbit. The total ΔV required was dependent on several factors:

- Launch site speed
- Orbit speed
- Losses (gravity and drag)
- Orbit transfer speeds

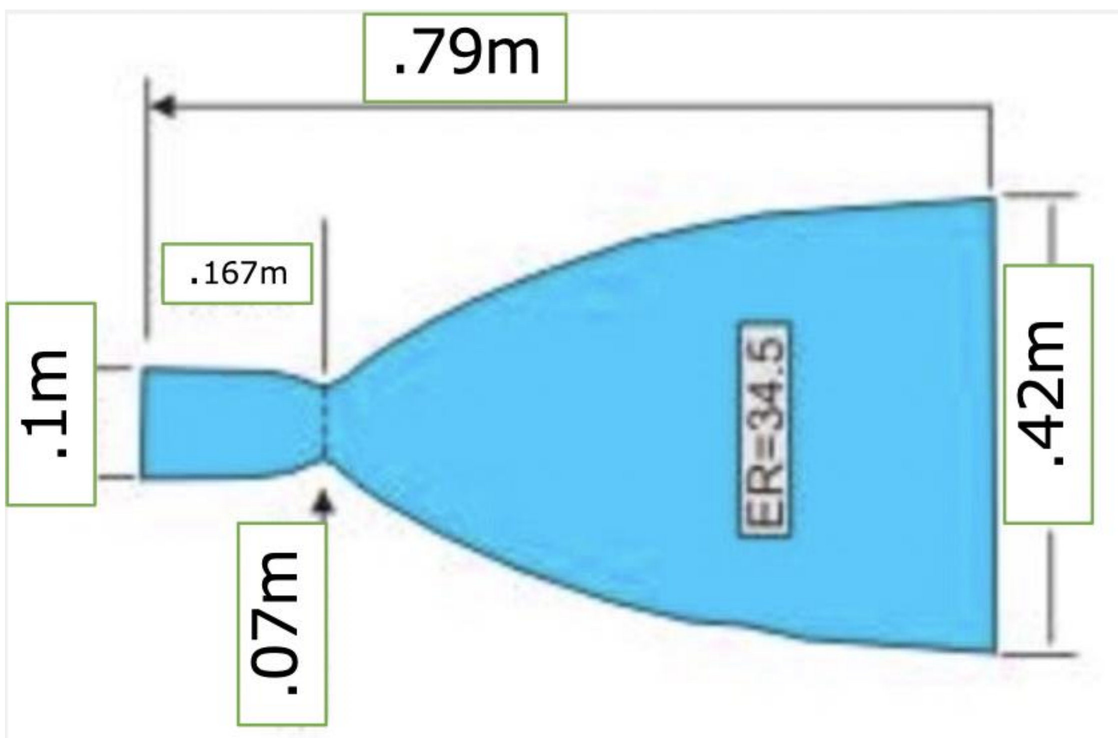
Using the launch site coordinates, the v_{LS} (Earth velocity at the launch site) could be calculated. The orbit and transfer speeds were calculated using the Vis-Viva equation. Finally, the gravity loss was calculated using the conservation of energy equations with a knockdown factor of 80% while the drag was an assumed 200 m/s. The total ΔV of the mission equaled 9917 m/s.

Design

To reduce to the cost of the program and maximize performance, the Venetia LV would utilize methalox (Liquid Oxygen/Liquid Methane) as its propellant for both its 1st and 2nd steps. As of May 2022, there have been no methalox rockets flown into space. However, SpaceX as well as a handful of other companies have begun testing and developing thrusters that use methalox. It was determined that SpaceX's Raptor engine would be used as the LV's thrusters for both steps due to its great performance. The SpaceX Raptor engine provides great structural coefficients in the atmosphere as well as in the vacuum of space. It provides an I_{sp} (specific impulse) of 330s and an I_{sp} of 380s in a vacuum, with an effective I_{sp} of 363s. Although the structural coefficients could be calculated using the data provided by SpaceX, their thrust was scaled down by a factor of about 10 to work with the smaller Venetia LV since the Raptor engines were originally optimized for the Starship LV. As a result, the size of the engines was also scaled down. The adjusted engine specifications table and engine figure are below:

Scaled to Designed LV

Step 1		**Step 2**	
Thrust (N)	185089.54	Thrust (N)	54155.75
Exit Area (m^2)	0.14	Area (m^2)	0.14
Radius (m)	0.21	Radius (m)	0.21
Exit Diameter (m)	0.42	Diameter (m)	0.42
Height (m)	0.79	Height (m)	0.79



With the adjusted structural coefficients and I_{sp}'s of each step, the payload mass, and the total ΔV, it was then possible to solve for the structural mass and propellant mass of each step using the ideal rocket equation. Since the LV is a TSTO, the ΔV had to be split between the first and second steps. In addition, it was also necessary to determine whether the losses due to gravity and drag would all be charged to the first step or if it would be smeared across both steps. It was determined after analysis that the losses should all be charged to the first step to maximize performance. By the time the second step thrusters would ignite, it could also be assumed that the losses due to drag and gravity would be negligible. To identify the best way to split the total

ΔV, a chart identifying different percentages of step 1 and step 2 ΔV and their respective masses was generated. The chart is below:

Case	1	2	3	4	5	6	7
Stage 1 deltaV_1	30%	35%	40%	45%	50%	55%	60%
Stage 2 delta_V2	70%	65%	60%	55%	50%	45%	40%
required delta_V2 (m/s)	5710.2	5302.3	4894.4	4486.6	4078.7	3670.8	3263.0
stage 2's loss (m/s)	0.0	0.0	0.0	0.0	0.0	0.0	0.0
mass ratio u2	4.6	4.1	3.7	3.3	3.0	2.7	2.4
mp2 (kg)	-51345.9	10809.3	4415.2	2573.6	1702.8	1197.7	869.6
ms2 (kg)	-14524.4	3057.7	1248.9	728.0	481.7	338.8	246.0
m02 (kg)	-65495.3	14242.0	6039.2	3676.6	2559.5	1911.6	1490.5
required delta_V1 (m/s)	4467.8	4875.7	5283.6	5691.4	6099.3	6507.2	6915.0
stage 1's loss (m/s)	1759.6	1759.6	1759.6	1759.6	1759.6	1759.6	1759.6
staging speed (km)	2.7	3.1	3.5	3.9	4.3	4.7	5.1
mass ratio u1	3.5	3.9	4.4	4.9	5.5	6.2	7.0
mp1 (kg)	-226383.7	61555.2	32893.5	25572.1	23210.7	23348.5	25863.1
ms1 (kg)	-24911.7	6773.6	3619.7	2814.0	2554.2	2569.3	2846.0
m01 (kg)	-250920.4	68703.8	36888.1	28761.1	26139.9	26292.8	29084.1
m00 (with PL) [kg]	**-316790.7**	**82570.8**	**42552.3**	**32062.7**	**28324.4**	**27829.3**	**30199.7**

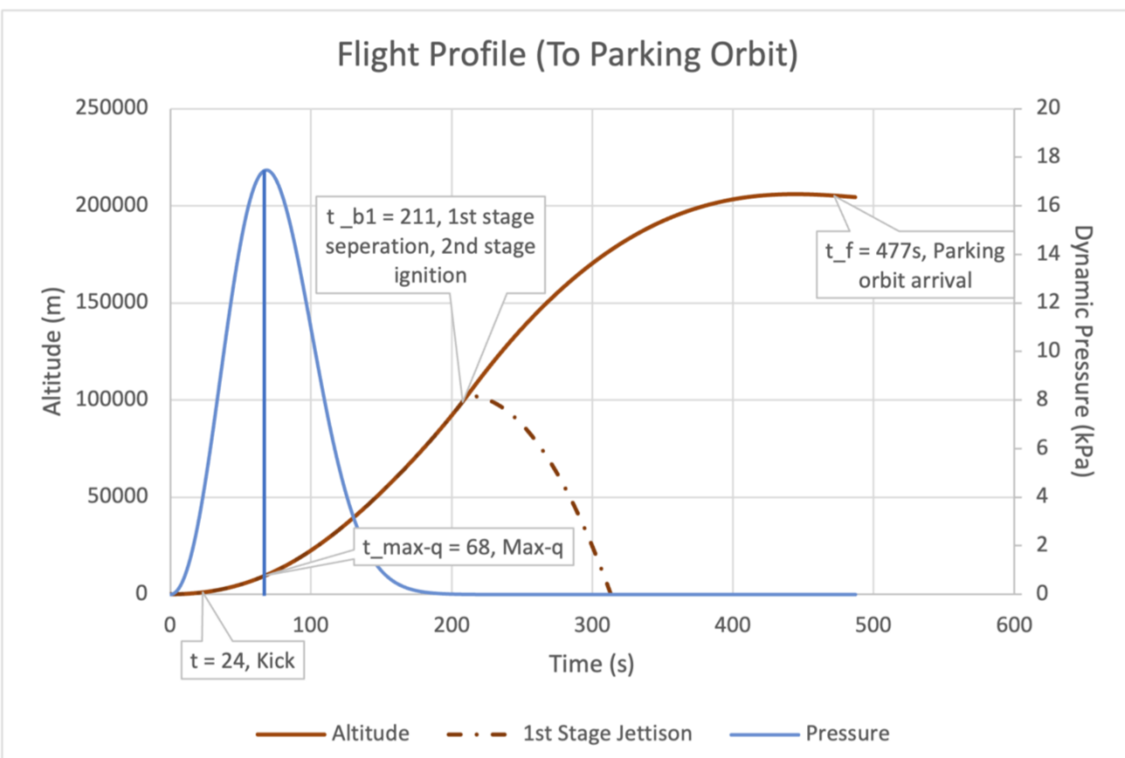
It should be noted that the best choice would be the percentage combination that offered the lowest m_{00} (gross liftoff mass) [GLOM]. Originally, the best combination was 40% and 60% of total ΔV would be charged to the first and second steps, respectively. However, due to the adjustment of the structural mass coefficients, the best combination was 55% and 45% for the first and second steps, respectively. The error was not noticed until the writing of this report. The only difference that this error caused is that the proposed LV is much heavier and, as a result, more expensive than what it should be.

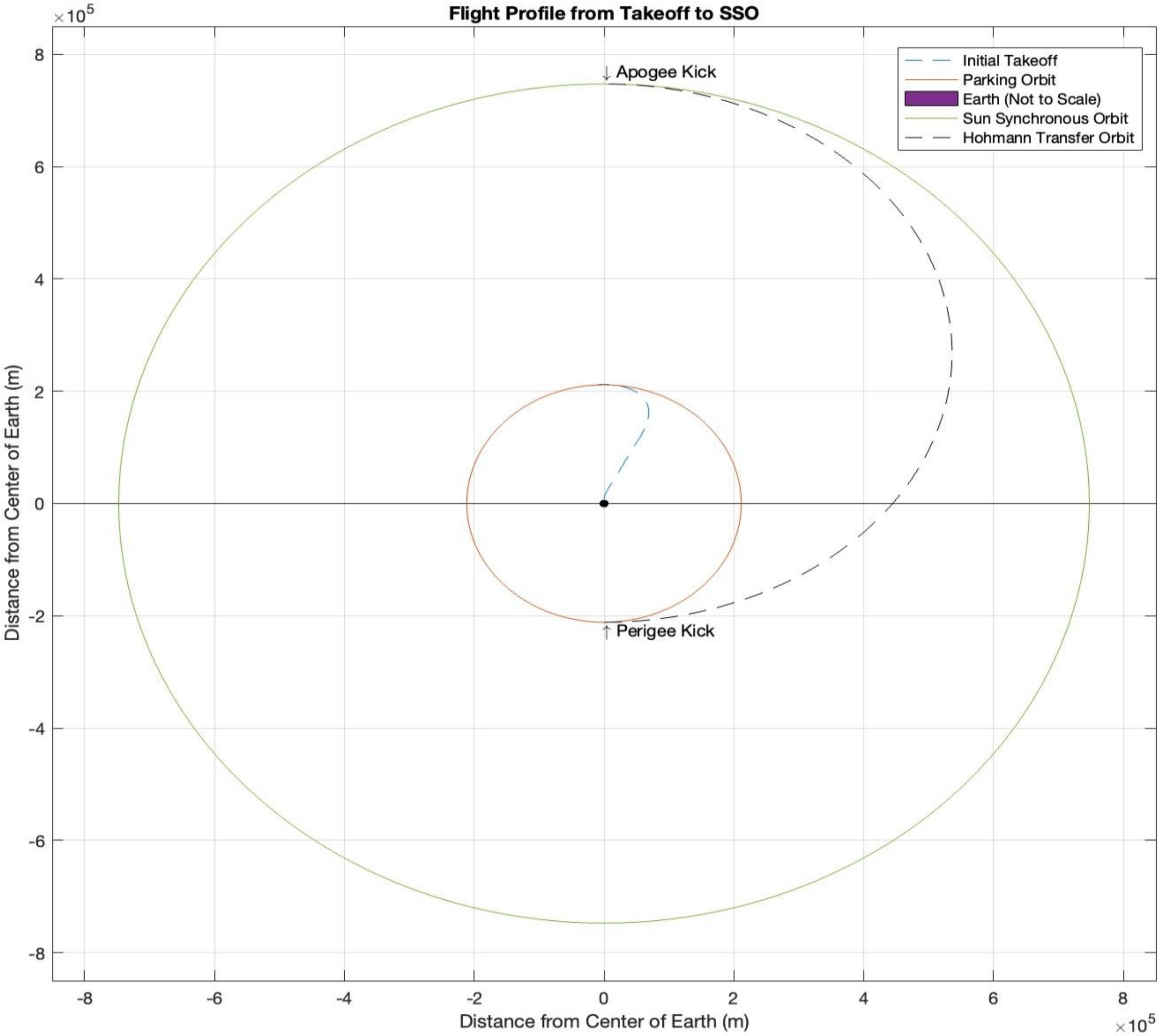
CONOPS / Flight Profile

A table describing the vehicle characteristics is below:

	Step 1	Step 2
Gross Mass (T)	Liftoff Mass = 42.56	Step 2 Ignition Mass = 6.03
Initial T/W	1.33	0.95
Total Thrust (kN)	555.3	54.2
Propellants (fuel/ox)	Methane/LOx	Methane/LOx
Specific Impulse (s)	330, 380, 363	380
Structural fractions	0.0991	0.2205
Speed after losses (km/s)	staging = 3.5	shutdown = 8.16
deltaV (m/s)	5283.2	4894.4
Losses (m/s)	1759.6	0
V_LS (m/s)	261	0
Propellant Mass (T)	32.9	4.4
Structural (Dry) Mass (T)	3.6	1.3
Length (m)	17.4	10.4

Plots simulating the parking orbit flight profile and the full flight profile are below:



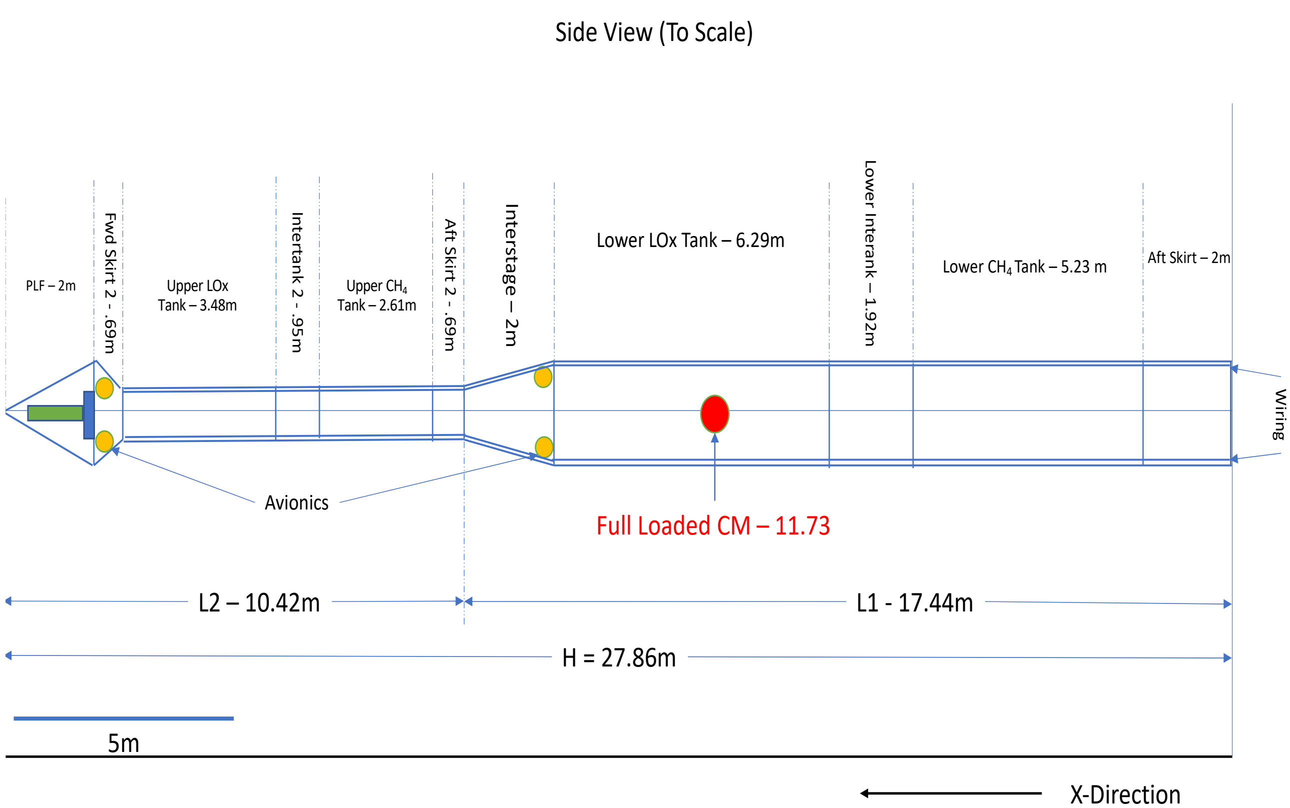


The Venetia LV uses the 3 scaled down Raptor engines to provide an initial thrust to weight ratio (T/W) of 1.33. About 24 seconds into the flight at an altitude of approximately 1km, a flight path "kick" of about 1.98° occurs to ensure that the vehicle gradually becomes perpendicular to the Earth by the time it reaches its parking orbit. At approximately 68 seconds and at an altitude of 9.8km, the vehicle experiences maximum dynamic pressure, Max-q. At 211 seconds, the 1st stage separates, and the 2nd stage ignition occurs. It is assumed that the 2nd stage ignites immediately after 1st stage jettison and that no loss in velocity occurs. Finally, the LV arrives at a circular parking orbit of 205km above the Earth's surface 477 seconds into its flight with a velocity of 7500 m/s (the ΔV required at this altitude is 7523 m/s). The 2nd stage remains in this parking orbit, without consuming propellant, until a desired perigee location arrives. Part of the remaining propellant is then used when the LV arrives at said position and a ΔV "perigee kick" of about 150 m/s occurs to propel the vehicle into an elliptical transfer orbit to arrive at its final altitude destination of 741km. The transfer orbit lasts for about 2820.25 seconds, or 47 minutes. Once the LV reaches the final altitude of 741km, a final ΔV "apogee kick" is done using the remaining propellant to circularize the orbit. The 2nd step is then jettisoned and the satellite orbits freely around the Earth.

It should be noted that the Earth is not drawn to scale in the Full Flight profile since it would not effectively and dramatically illustrate the LV's path. It is scaled down about 1000 times its actual size (Earth's radius is ~6400km and the parking orbit is only 205km above Earth's surface). The initial takeoff in the Full Flight profile also does not accurately portray the true initial flight due to conversions from cartesian to polar coordinates. The LV travels downrange for the entirety of the initial takeoff, which covers about 4% of Earth's circumference. **To better understand the entire flight profile, click the link below to watch the simulated graph:**

Full Flight Profile Live Simulation

A side view of the LV is provided below:



It is important to note that the engines are not pictured in the side view, however, they were accounted for when calculating forces, moment, mass, etc. There are 3 scaled down Raptor engines at the bottom of the 1st step and 1 scaled down Raptor engine at the bottom of the 2nd step, housed in the interstage.

III. Inboard Profile: Mass Properties

Vehicle Sizing

The vehicle was then sized based on the structural and propellant masses that were calculated from the Design section. The design of the LV was also limited with a minimum diameter of 2m and a fineness ratio of 2.5. The tank masses and lengths were calculated by using the densities of the fuel and oxidizer to calculate the volume necessary to store them. The individual masses of the oxidizer and fuel for each step were calculated by using the nominal mixture ratio of the Raptor engines. Ullage and residue propellant mass were also taken into account for the propellant mass calculations. The remaining masses and lengths were calculated using the Mass Estimating Relationships (MERs) found in the book. Since the there is no public data for insulation mass for liquid methane, the insulation mass was calculated using liquid hydrogen data. The size of the thrusters was sized down from the actual sizes of the Raptor engines to accommodate for the smaller thrust requirements. The first step uses 3 engines while the second step uses 1. Although the second step thrust requirement is lower than what is provided by a Raptor engine, the engine would be throttled to 30% of maximum thrust. The final mass estimation resulted in a negative margin of 7kg. In other words, the estimated GLOM was 7kg lighter than the estimated GLOM from the Design section. The LV also had a fineness ratio greater than 2.5.

As can be seen from the side view of the LV in the CONOPS subsection, the LV is comprised of 3 main sections: the payload fairing, the 2nd step, and the 1st step. To minimize complexity of the LV, the diameter of the 2nd step was reduced to 1m and expands back to 2m for the required payload dimensions by way of a forward aft skirt. The design of the LV could now continue to Center of Mass (CM) and Moment of Inertia (MoI) calculations.

CM and MoI in Pitch and Roll Calculations

To get an idea how the LV will move as it flies through the atmosphere and how different loads will affect its integrity, the CM and MoI were calculated for the different components of the LV by using the sizing and masses from the Vehicle Sizing section. The results are below:

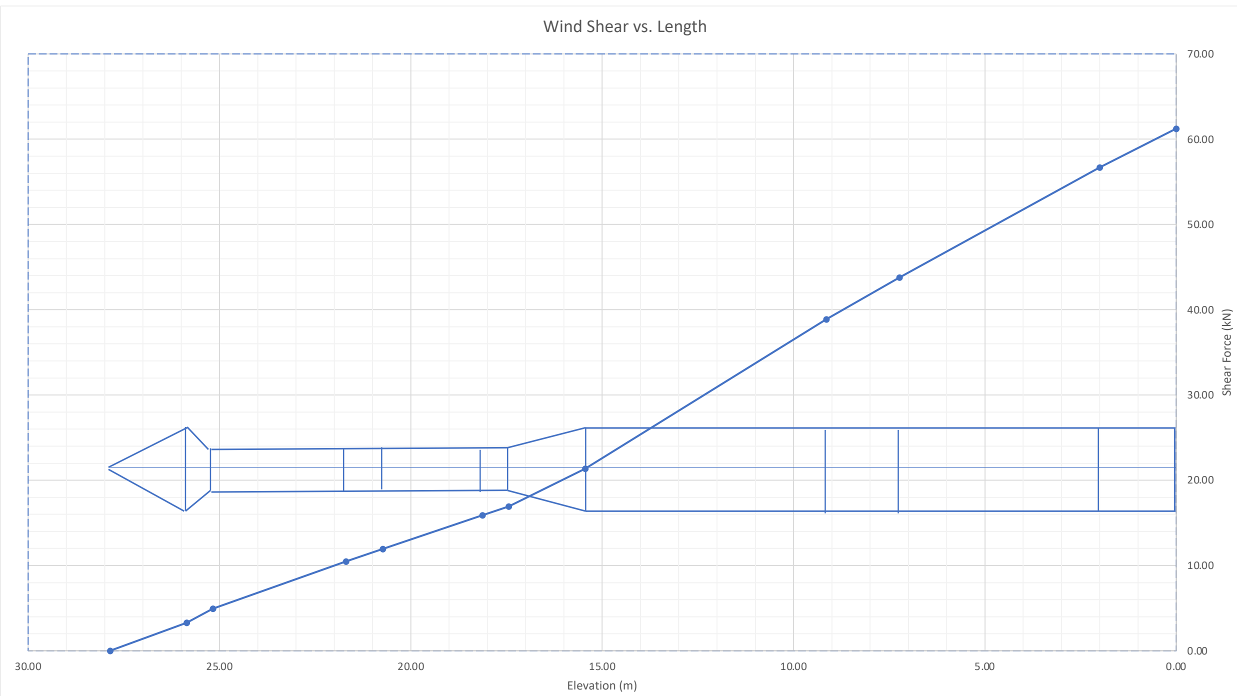
Item	Mass (kg)	Ref. Distance (m)	Moment (kg*m)	Distance from CM	kg*m^2 (CM)	J_pitch	J_roll
Payload Fairing	118.18	27.19	3213.77	15.46	28264.61	28264.61	59.09
Payload + PAF	453.31	27.36	12402.78	15.63	110765.61	110765.61	113.33
Forward Skirt 2	53.25	25.86	1377.02	14.13	10633.80	10633.80	33.28
Avionics 2	136.11	25.84	3517.05	14.11	27103.38	27103.38	34.03
Wiring 2	88.29	22.65	2000.00	10.92	10534.11	10534.11	22.07
Ox Tank 2	37.89	23.43	887.94	11.70	5191.09	5191.09	9.47
Ox Insulation 2	15.14	23.43	354.84	11.70	2074.45	2074.45	3.79
LOx Residual 2	85.60	21.34	1826.68	9.61	7906.77	7906.77	0.00
Intertank 2	39.99	21.21	848.39	9.49	3598.41	3598.41	10.00
Fuel Tank 2	29.40	19.43	571.23	7.70	1744.66	1744.66	7.35
Fuel Insulation 2	12.06	19.43	234.35	7.70	715.75	715.75	3.01
Fuel Residual 2	24.81	17.78	441.05	6.05	907.34	907.34	0.00
Aft Skirt 2	43.47	18.13	788.14	6.40	1781.08	1781.08	10.87
Thrust Structure 2	141.59	17.51	2478.60	5.78	4724.48	4724.48	35.40
Gimbals 2	6.23	17.51	109.13	5.78	208.01	208.01	0.00
Engines 2	332.17	17.57	5835.48	5.84	11324.62	11324.62	0.00
Interstage	129.21	16.55	2138.86	4.82	3008.02	3008.02	80.75
Avionics 1	136.11	16.11	2192.60	4.38	2611.81	2611.81	136.11
Wiring 1	147.83	8.72	1289.30	-3.01	1337.09	1337.09	147.83
Ox Tank 1	277.62	12.29	3413.34	0.57	88.98	88.98	277.62
Ox Insulation 1	55.88	12.29	686.99	0.57	17.91	17.91	55.88
LOx Residual 1	627.17	8.44	5293.31	-3.29	6783.24	6783.24	0.00
Intertank 1	159.96	7.23	1157.00	-4.50	3233.25	3233.25	159.96
Fuel Tank 1	235.92	4.62	1089.13	-7.11	11934.01	11934.01	235.92
Fuel Insulation 1	48.38	4.62	223.34	-7.11	2447.18	2447.18	48.38
Fuel Residual 1	195.38	1.29	252.61	-10.44	21278.13	21278.13	0.00
Aft Skirt 1	167.13	1.00	167.13	-10.73	19237.88	19237.88	167.13
Thrust Structure 1	424.78	0.13	53.10	-11.60	57194.95	57194.95	424.78
Gimbals 1	18.70	0.13	2.34	-11.60	2518.22	2518.22	0.00
Engines 1	996.52	0.25	249.13	-11.48	131302.22	131302.22	0.00
Propellant Y(1)/N(0	1	1					
LOx 2	3424.04	23.43	80236.37	11.70	469080.61	469080.61	0.00
Fuel 2	992.47	19.43	19286.50	7.70	58905.21	58905.21	0.00
LOx 1	25086.78	12.29	308438.34	0.57	8040.64	8040.64	0.00
Fuel 1	7815.20	4.62	36078.45	-7.11	395326.80	395326.80	0.00
Totals	**42556.59**	**kg**	**499134.26**		**1421824.35**	1421824.35	**2076.06**
CM Location	**11.73**	**m**					

IV. Ground Loads

To choose a material that would be the body of the LV, it was necessary to calculate the different loads acting on the body of the vehicle at the ground level: maximum shear force, maximum bending moment, and the maximum axial load. Using the previously calculated CM and MoI, those loads can be calculated.

Maximum Shear

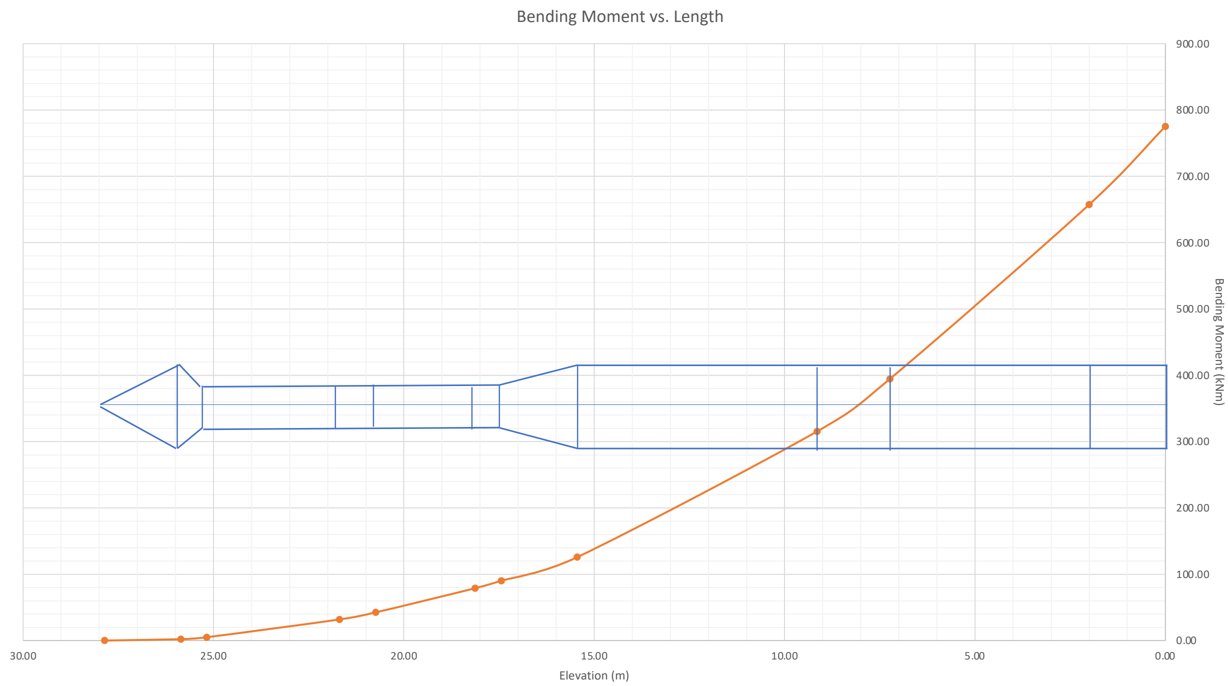
The shear force, or the horizontal forces acting on the vertical sections of the LV to the right and left of that section, were calculated using the methods described in the book. Wind loads were the primary forces acting on the LV. The results are below:



Note that the sum of the of the shear forces are maximum at the bottom of the LV and no forces are present at the top of the vehicle.

Maximum Bending

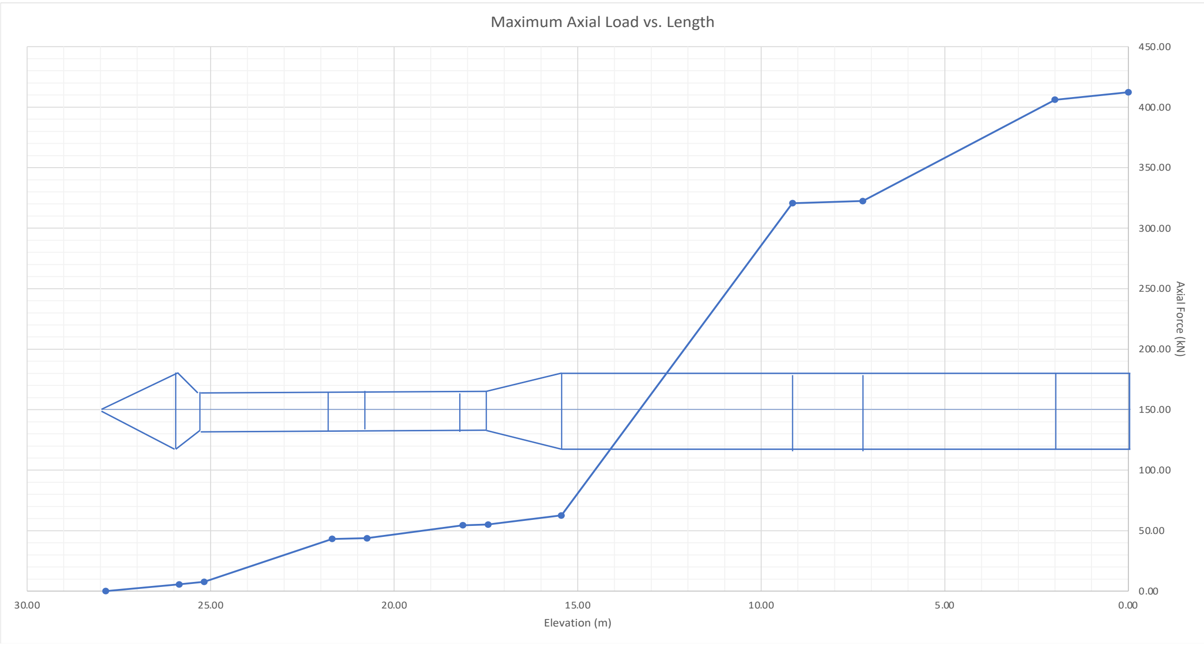
The maximum bending moment is the moment of the shear force acting on the vertical sections of the LV to the right and the left. In other words, it is a measurement of a force a given distance away from the section's CM. The results are below:



Note that the sum of the of the bending moments are maximum at the bottom of the LV and no forces are present at the top of the vehicle.

Maximum Axial

The maximum axial load is the vertical force that a section exudes on the bottom section. The results are below:



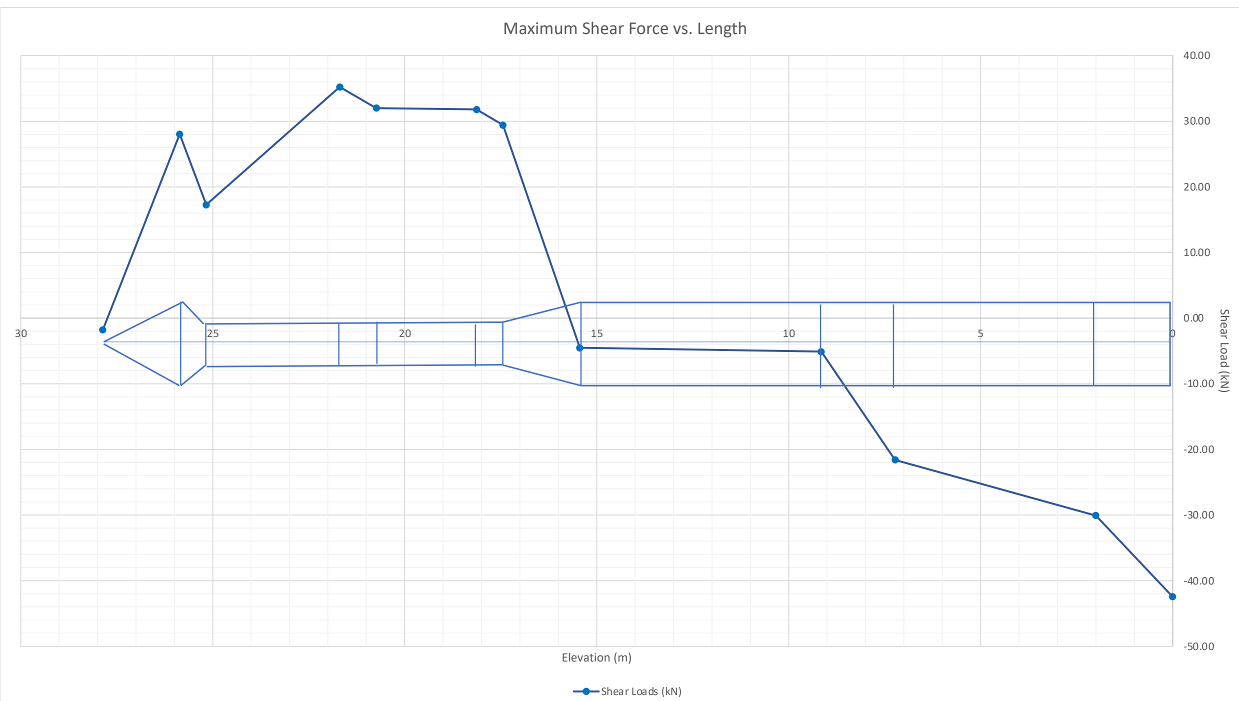
Note that the sum of the axial loads is maximum at the bottom of the vehicle. The sum of the axial loads should also equal the gross liftoff weight (GLOM × g_0). In this case, the sum of the axial loads is within 5% of the gross liftoff weight.

V. *Max-q Flight Loads*

Similar to the Ground Loads section, the loads that the LV experiences when in flight is also crucial when determining the material for the body of the LV. The maximum loads that the LV experiences occurs when the vehicle is under the most pressure, also known as Max-q. Using the flight profile calculations from the Flight Profile subsection, it is possible to know at what time Max-q occurs, how much pressure the vehicle is under, how much propellant is consumed, and other important parameters. After determining how much propellant is consumed, the CM and MoI were recalculated to get the accurate CM and MoI at Max-q. Those new values were used to calculate the Max-q Flight Loads.

Maximum Shear

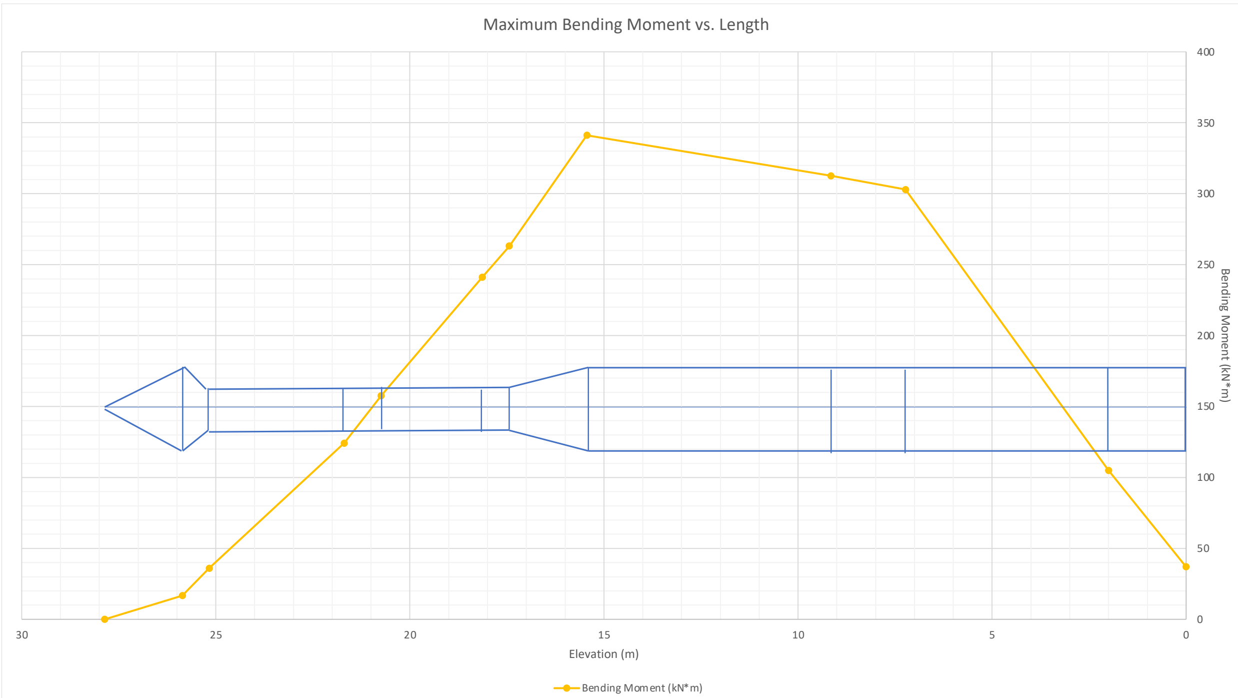
The results of the calculations are below:



Note that the sum of the shear force is approximately T × sin(δ_{TVC}) at the thruster gimble block. In other words, it is the horizontal force enacted on the aft of the vehicle caused by the angle of the thrusters.

Maximum Bending

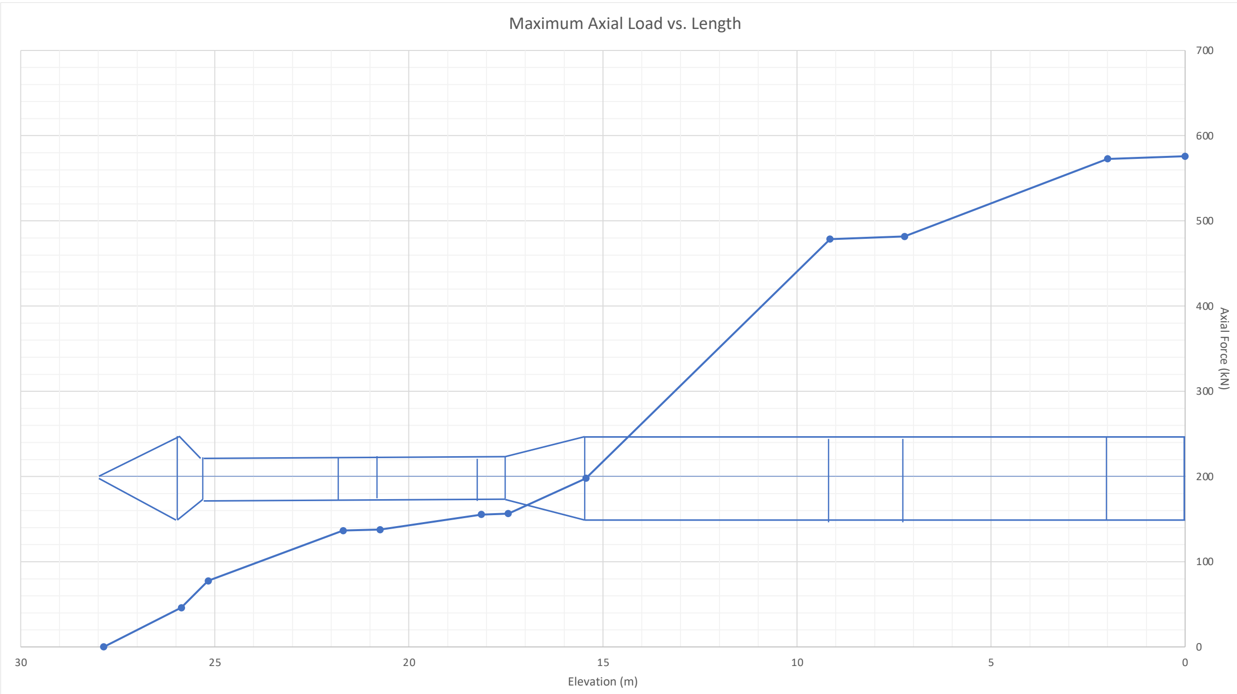
The results of the calculations are below:



Note that the sum of the bending moments should be zero at both ends of the LV. In these calculations, the bending moment at the end of the vehicle is not zero but it is within an acceptable range.

Maximum Axial

The results of the calculations are below:



Note that the sum of the axial loads should equal T × cos(δ_{TVC}). However, since δ_{TVC} is a small angle, cos(δ_{TVC}) is approximately equal to 1. Therefore, the sum of the axial loads at the end of the LV is the total force that the thrusters are generating. The calculations here are in agreement.

VI. Stress Analysis

Material Selection

Since it was assumed that the propellant tanks would also be used as the structure of the vehicle, two different materials were needed. For intertanks and interstages, the most common material used is aluminum 7075-T6 and, as such was selected as a material. A list of common liquid propellant tank materials was provided, and it was determined that aluminum 2219-T87 would be used. Their material properties are listed below:

Material Description			
Aluminum 2219-T87	**Notes**	**Aluminum 7075-T6**	**Notes**
Operating Temp. (K)	90-111	Operating Temp. (K)	293
Density (kg/m^3)	2840	Density (kg/m^3)	2810
Elastic Modulus (MPa)	73100	Elastic Modulus (MPa)	71700
Yield Stress (MPa)	393	Yield Stress (MPa)	503
Source: Referenced on 4/4/22 https://asm.matweb.com/search/SpecificMaterial.asp?bassnum=MA2219T87		Source: Referenced on 4/4/22 https://asm.matweb.com/search/SpecificMaterial.asp?bassnum=ma7075t6	

Stress calculations provided by the book, the maximum ground and Max-q stresses could be calculated. The stresses were then compared to the material properties to ensure that the materials were safe to use.

Maximum Ground and Max-q Stress

The three locations that experienced the highest bending moment for the ground loads and the three locations with the highest bending moment at max-q were chosen for the six maximum stress calculations. The results are below:

Location	Material	Thickness (mm)	Load Case	Calc. Stress (MPa)	Yield Stress (MPa)	MS (%)
Lower Intertank	Al 7075-T6	0.75	Max-q	402.40	503	25
Lower Aft Skirt	Al 7075-T6	0.85	Ground	402.40	503	25
Bottom	Al 7075-T6	0.97	Ground	402.40	503	25
Lower Fuel Tank	Al 2219-T87	1.03	Ground	314.40	393	25
Lower Fuel Tank	Al 2219-T87	1.09	Max-q	314.40	393	25
Lower Ox Tank	Al 2219-T87	1.38	Max-q	314.40	393	25

The results were all calculated by using a minimum measure of safety (MS) of 25%. The calculations also allowed to find the thickness of each component to get an updated mass estimation by using the materials density.

Updated Mass Estimation

The updated mass estimation table of the six locations is below:

Location	MER Mass (kg)	New Mass (kg)	Margin	Percent Difference
Lower Intertank	159.96	25.54	134.42	84%
Lower Aft Skirt	167.13	30.04	137.09	82%
Bottom	167.13	30.04	137.09	82%
Lower Fuel Tank	235.92	96.30	139.62	59%
Lower Fuel Tank	235.92	101.48	134.44	57%
Lower Ox Tank	277.62	155.33	122.29	44%

It is clear that there is a significant difference between the MER calculated mass and the new updated mass. As a result, the vehicle will weigh a lot lighter than previously predicted. This provides several advantages since a bigger payload can be added (while respecting the staging effect) and larger missions can be accomplished if necessary. However, for the scope of the Venetia program, it is a significant waste in money since there a lot of propellant will possibly not be used. The newly updated mass estimation also does not take into account any support beams or struts that would be added.

VII. Recovery of 1st Step of LV

Similar to SpaceX's Falcon 9, the Venetia LV will also attempt to recover its 1st step to save on costs. Several assumptions were made to simplify calculations:

- 1st step engines have a maximum TVC deflection of $\delta_{\max}= \pm 6°$
- Engine thrust can change instantaneously
- Thrust is magically set to T = 1.5 ($m_{1f} \times g_0$) on approach
- 1st step m_{1f}, MoI, and CM remain constant
- Aerodynamic forces and wind gusts are ignored

Using the CM and MoI spreadsheet from the Inboard Profile section, the dry mass (m_{1f}) of the 1st step only along with its CM and MoI can be calculated. To calculate the time it takes for the 1st step to reach zero rotational displacement and zero rotational rate to prevent tipping, it was necessary to understand the rotational dynamics of the system and utilize the equation for applied moment, $M = J\theta''$ (applied moment = inertia × angular acceleration). It was determined that it would take 6.86s.

VIII. *Cost Estimation*

The final step of the design process is the cost estimation. Using mission parameters such as propellant type, propellant weight, dry mass, etc. as inputs, the overall cost of the program can be calculated. A 1965 U.S. Air Force Space Planner Guide provides several models that these inputs can be used for to estimate the cost. The final cost of the program is based on 5 key cost components:

- Total Development Test & Evaluation (DT&E) [$938.6M]
- Total Facilities Costs [$125.8M]
- Total Aerospace Ground Equipment [$16.8M]
- Total Hardware Production [$247.8M]
- Total Operation Costs [$244.5M]

It is important to note that during cost calculations, several assumptions were made for the sake of mirroring modern day programs. It was assumed that there would only be 2 vehicles for launch vehicle testing and 2 vehicles for space vehicle test flights. 6 vehicles would be launched every year from 1 specific launch site with a vehicle production rate of 1 per month. Finally, the program would be operational for 5 years and have a total of 24 operational launch vehicles.

The final total cost of the program was $1.6B in 1965 USD. After adjusting for inflation, the program would cost $14.4B in 2022 USD. To put into perspective, the U.S. spent $25.8B (over $250B in 2022 USD) between 1960 and 1973 on the Apollo project. It is possible that the number calculated is an overestimate. Since Methalox is a new type of propellant, the Space Planner Guide does not provide data for it. Instead, the LH_2 (Liquid Hydrogen) model was used. Methalox is cheaper than most fuels and it is possible that a more accurate model could prove cheaper costs. As can be seen from the Updated Mass Estimation subsection, the new predicted dry mass of certain LV components is considerably lighter than the original prediction. A cheaper final total cost can also be achieved if a thorough investigation of the new estimated mass is conducted.

IX. Conclusion / Lessons Learned

Based on the design and mathematical simulation of the LV, the vehicle proposition would pass the initial development stage. The flight profile simulation demonstrates that the LV will reach the parking orbit with the required ΔV to then continue to the transfer and final orbits. Load and stress calculations indicated that the vehicle could handle the maximum loads and stresses that it would experience during flight. The final mass estimation also indicated that the vehicle could have better performance and/or be cheaper than originally predicted. The final cost of the LV program is comparable to other programs, a finding that is significant considering the LV is heavier than it should be. This report would be eligible to be presented and possibly moved to the next phase of development.

As an electrical engineering undergraduate, it was quite the hurdle to learn a different field of engineering and keep up with the pace of the class. From understanding mechanics to researching different material properties, each assignment leading up to the final project offered a challenge. However, it was well worth the effort.

In terms of completing the assignments, not many problems were encountered along the way. I did have several errors unfortunately. During the initial design portion of the project, I neglected to check to see if my LV would produce enough thrust to get a T/W above 1. By the time I reached a later stage, I realized that my LV would not liftoff and I had to restart the design process, which led to me accidentally picking a heavier vehicle than required. Although minimal here, in the real world, an error like that can cost millions of dollars! Another error occurred during the flight profile calculations. I was solving for a flight path "kick" to reach a circular orbit at 35km instead of the 200km parking orbit necessary. Fortunately, it was far enough in the design that minimal recalculations were required. In the future, it will be necessary to conduct several calculation checks to ensure that errors are minimized. In the next iteration of the project, I would like to go back and recalculate and re-simulate the proposed LV design using the lighter vehicle choice and compare the final cost of that version to see how much that error would cost. A stability test is the only thing lacking from the final report. In the next iteration, I would also like to complete that.