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Abstract

The Moon-based Advanced Reusable Transportation Architecture (MARTA) Project conducted an in-depth investigation of possible Low Earth Orbit (LEO) to lunar surface transportation systems capable of sending both astronauts and large masses of cargo to the Moon and back. The goal of this project was to create a profitable venture with an Internal Rate of Return (IRR) of 25%.

The architecture was quickly narrowed down to a traditional chemical rocket using liquid oxygen and liquid hydrogen. However, three additional technologies identified as potentially cost saving were: aerobraking, in-situ resource utilization (ISRU), and a mass driver on the lunar surface.

The vehicle was modeled using the Simulated Probabilistic Parametric Lunar Architecture Tool (SPPLAT) that incorporated several different engineering disciplines. This tool uses ISRU propellant cost, a dry weight reduction due to improved materials technology, and vehicle engine specific impulse as inputs and provides vehicle dry weight, total propellant used per trip, and price to charge the customer in order to guarantee an IRR of 25% as outputs. Estimation error, market growth, and launch cost uncertainty were also considered.

The results of the project show that the desired operation is possible using current technology. Based on the stipulation that the venture be profitable, the price to charge the customer was highly dependent on ISRU propellant cost and relatively insensitive to the other inputs. With the best estimate of ISRU cost set at $1000/kg, the resulting price to charge the customer was $2600/kg of payload from LEO to the lunar surface. If ISRU cost can be reduced to $160/kg, the price to the customer is reduced to just $800/kg of payload. Additionally, the mass driver only proved to be cost effective at an ISRU propellant cost greater than $250/kg, although it reduced total propellant used by 35%.

Nomenclature

EOI	Earth Orbit Insertion
ERO	Elliptical Refueling Orbit
ISRU	In-situ Resource Utilization
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
	Lunar Lander and Transfer Vehicle
MARTA	Moon-based Advanced Reusable Transportation Architecture
MT	Metric Ton
NAFCOM96	1996 NASA Air Force Cost Model
RFP	Request For Proposals
RSE	Response Surface Equation
RSM	Response Surface Methodology
SPPLAT	Simulated Probabilistic Parametric Lunar Architecture Tool
TEI	Trans-Earth Injection
TLI	Translunar Injection
TRF	Technology Reduction Factor
WBS	Weight Breakdown Statement
WAF	Weight Adjustment Factor

Introduction

More than thirty years after Neil Armstrong first walked on the Moon, the scientific community is experiencing a renewed interest in Earth's only natural satellite. The recent Clementine and Lunar Prospector missions have revealed that there is still much more to discover about the Moon[1]. These discoveries have led small companies like Orbital Technologies to complete studies in attempts to verify that ice exists at each of the Moon's two polar regions. At the same time, groups like the Artemis Society International are advocating the

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establishment of privately financed permanent human colonies on the Moon for the sole purpose of making a profit[2].

While seemingly unrelated, each of these lunar missions has a single unifying feature. They all are dependent on the construction and operation of a commercially viable Earth-Moon transportation system. Considering the declining budgets approved each year for the National Aeronautics and Space Administration (NASA), the government will not be able to fund a transportation system of the type that is needed. Instead the financial backing for the program must come from private industry. Since the driving force behind any private industry venture is profit, there must be a level of return on the investment commensurate with the risk involved in developing such a transportation system.

The need for an Earth-Moon transportation system combined with the financial requirement that the system be profitable was the impetus for designing a Moon-based Advanced Reusable Transportation Architecture (The MARTA Project). The main goal of the project was to design a transportation system capable of moving astronauts and large amounts of cargo between a space station in Low Earth Orbit (LEO) and the lunar surface.

The main mission requirements assumed for this study are as follows:

1) 10 flights/year of 20 MT cargo
2) 5 flights/year of 40 MT cargo
3) 3 flights/year of 60 MT cargo
4) 4 manned flights/year of 5 astronauts
5) Half of all cargo and astronauts are delivered to a polar base and the other half to an equatorial base
6) Cargo must be delivered to the Moon within 4 weeks of launch from the Earth
7) Manned missions must not take longer than 5 days in transit

Additional requirements for the project are that all of the astronauts taken to the Moon must be returned to LEO, and the return cargo load is half the size of the outbound cargo load. Annual market growth is expected to be 5%, but could range from 0% to 15%. To offset startup costs, it was assumed that NASA would contribute 50% of the money required for Design, Development, Testing, and Evaluation (DDT&E) of the system and would be a guaranteed customer for seventeen years after the initial year of operation (2018). A final requirement for a successful design was that a private company that undertakes the development of the system would be able to make a 25% rate of return on their initial investment over the life of the project.

Earth to Moon Transportation Architecture Selection Process

To minimize the possibility of overlooking a potential solution, the design team entered the process without preconceived notions regarding the final architecture. As such, it was difficult to narrow down an essentially infinite design space to a single architecture. The only insight the design team had into the problem before the brainstorming session was that the propellant usage of the system needed to be minimized if the operation was to be profitable. This fact came from a preliminary economic analysis that indicated the largest overall costs associated with the Earth-Moon transportation system were operations costs. For an in-space system like this one, operations cost translates almost directly into propellant cost. Thus, going into the brainstorming session, the team knew that reducing the propellant usage was a necessity. After brainstorming, the following four architectures were identified as most promising: a momentum-transfer tether system, a nuclear thermal rocket system, an electric propulsion system, and a chemical liquid rocket engine combined with an in-situ resource utilization (ISRU) program to provide propellant. Representative images of each of these systems appear below as Figure 1. The figure shows (from left to right) a satellite accelerating via a momentum-transfer tether, a nuclear thermal rocket engine, an electric rocket engine, and a chemical liquid rocket engine.

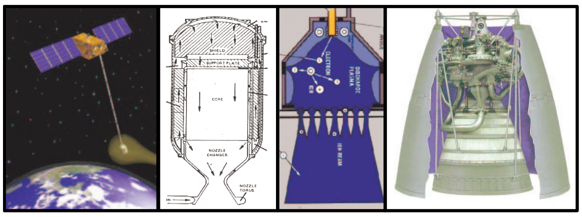

Figure 1: Propulsion Systems Considered

With these four systems identified, more detailed analyses provided a better idea of the main benefits each offered as well as the main drawbacks to the systems. The analyses also allowed for a systematic down-selection process that resulted in a single architecture. The results of the down-selection showed that the tether system was not safe enough to be used with a human system. The main reason for this decision was that if the spacecraft missed the tether, it would not be able to enter the required orbit and could jeopardize the lives of the astronauts on board. Nuclear thermal rockets were eliminated from consideration because the design team felt that the environmental lobby would not allow a nuclear reactor to orbit Earth on a regular basis while the third candidate, an electric propulsion system, was eliminated because of time considerations. The current state of the art in electric propulsion requires a three-month period to move a satellite from LEO to Geostationary Orbit (GEO). As such, it would take longer than the thirty days allowed to move a vehicle from LEO all the way to the Moon. This left the chemical liquid rocket system using lunar resources to produce propellants on the Moon. This architecture was attractive based on the fact it uses proven technology and with ISRU it has the potential to use relatively low cost propellants since the cost of launching propellant from the Earth would be prohibitive.

One piece of technology that was included in each of the proposed system architectures was the use of an aerobrake maneuver through Earth's atmosphere when returning from the Moon. This procedure is used to further minimize the propellant usage and decrease the associated costs. The aerobrake minimized propellant usage because without it, the vehicle would have to burn its engine to slow down enough to be captured in Earth orbit and dock with the station. Even though crewed missions would use the same vehicle as the cargo missions, safety dictates that the aerobraking procedure not be used on the astronaut transfer missions.

An additional method of reducing overall propellant use was the implementation of numerous fuel depots, including one in low lunar orbit (LLO), one in LEO, and several in intermediate elliptical refueling orbits (EROs). This option would allow for a smaller vehicle dry mass due to a smaller fuel capacity. However, as the vehicle dry mass was small compared to the payload mass, there was limited advantage to having more than one refueling stop. Thus, all the depots except for one in an ERO were eliminated. Additional analysis of the orbital mechanics of a depot in ERO showed that the depot's orbit would precess too much and would limit the launch opportunities to two per month. To maintain the usefulness of in-space refueling, a just-in-time refueling plan was developed. Using additional vehicles to carry the additional propellant only when it is needed, the orbital precession of a fuel depot was avoided, as the refueling vehicle would be sent only as needed.

Baseline Operations/Architecture

The baseline mission architecture consists of a MARTA operated facility at the Moon's South Pole which is both the center of overall operations as well as the location of the propellant production facility which makes liquid oxygen and liquid hydrogen from lunar water ice. The South Pole was chosen because the majority of the lunar ice is located there[1]. If necessary, a similar facility can be constructed at the North Pole. The system uses a combined lunar lander and transfer vehicle (LLTV) design that allows a single vehicle to take returning cargo or astronauts from the Moon to LEO as well as inbound cargo or astronauts from LEO to the lunar surface. This same vehicle design also functions as an in-space refueling vehicle during a transfer mission. MARTA maintains no infrastructure at the Moon's equator, but supplies transportation services to the NASA base located there.

Baseline Vehicle Description

The MARTA vehicle serves as both lunar lander and in-space transfer vehicle. It remains as one unit throughout the entire mission. The aerobrake is used to capture into Earth orbit in the cargo and refueling missions. For astronaut missions, the aerobrake is not used even though it is available. Instead, a propulsive burn is used to capture a vehicle carrying crew members. The low thrust requirement for lift off from the Moon enables the same engine to be used for launch, landing, and all in-space propulsive burns. A three-view of the baseline MARTA vehicle is shown in Figure 2.

The vehicle is designed to accommodate four different configurations as shown in Figure 3. Each of these different payloads is fitted in the payload compartment either while the MARTA vehicle is docked in LEO or is on the surface of the Moon.

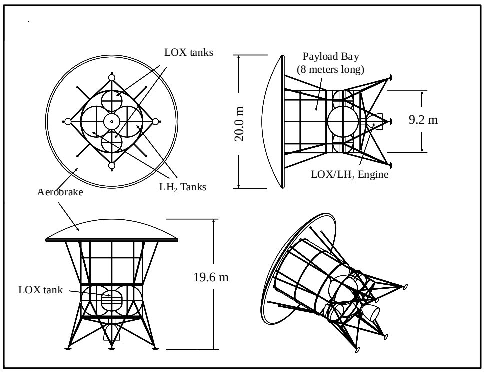

Figure 2: Three View of the MARTA Transfer Vehicle

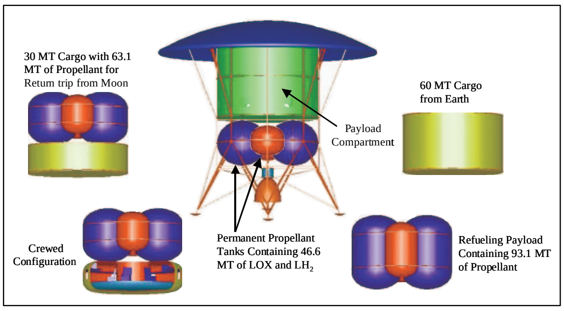

Figure 3: MARTA Transfer Vehicle with Available Payload Configurations

Due to the relatively low forces imposed on an in-space system, the vehicle itself is relatively light, as can be seen in the component weight breakdown given Table 1. The numbers in Table 1 apply to the vehicle regardless of the mission. Only the contents of the payload compartment change when the vehicle is outfitted for one of its various missions. Estimates show that the vehicle can expect to experience a maximum of 0.1 Earth g's during the aerobraking procedure and a maximum of 0.33 Earth g's during landing on the lunar surface. A finite element analysis shows that the truss structure designed for the vehicle is strong enough to withstand 1.5 Earth g's.

Table 1: Baseline Vehicle Weight Breakdown Statement

1.0	Body Group		1400 kg
1.1	Primary Structure	825 kg	
1.2	Thrust Structure	175 kg	
1.3	LOX Tank	150 kg	
1.4	LH2 Tank	250 kg	
2.0	Landing Gear		325 kg
3.0	LOX/LH2 Engine		325 kg
4.0	RCS Propulsion		125 kg
5.0	Aerobrake		1025 kg
6.0	Primary Power		1075 kg
7.0	Electrical Conversion and Distribution		400 kg
8.0	Environmental Control		375 kg
9.0	Avionics		375 kg
10.0	Margin		825 kg
	Dry Mass		**6250 kg**

In-Situ Resource Utilization (ISRU) Research

Human settlement of space must eventually involve the utilization of space resources. A key question is whether the use of such resources can be leveraged to reduce the costs and increase the profitability of near-term space development plans. An early application will most likely be space-based propellant production. While Earth-To-Orbit (ETO) launch costs remain high, use of space-based propellants looks promising. This is because the high cost of earth-based propellants allows even a relatively massive, inefficient space-based propellant manufacturing facility to be cost competitive. If ETO launch costs drop, the design requirements of an economically viable propellant manufacturing facility become more stringent.

Economics of Lunar Propellants

The team decided to investigate the use of lunar propellants in its lunar transportation architecture for two reasons. Initial economic assumptions made the use of Earth-based propellants financially impossible, so the only alternative, lunar propellants, had to be investigated. ETO launch costs were assumed to be $1600/kg of payload for a third generation reusable launch vehicle while payment for transporting payload from LEO to the lunar surface was initially targeted at $800/kg. Considering only propellant cost, it would have been necessary for each kilogram of propellant to transport two kilograms of payload from LEO to the lunar surface in order to break even. Such a high payload to propellant mass ratio (m_{PL}/m_p) is not feasible for near-term LTVs. In a Boeing study from 1993, a representative LTV traveling between LEO and LLO has a payload/propellant ratio of approximately one[3]. The baseline architecture presented here has a payload to propellant ratio of 0.26, largely because it acts as both a lunar surface lander and a transfer vehicle and must overcome the Moon's gravity. To break even just on the ETO cost of transporting propellant without considering investment and hardware procurement costs, the baseline architecture would need to charge $6000/kg to transport cargo from LEO to the lunar surface.

Lunar Polar Ice

The second reason for examining lunar propellant production was the new data available from the Clementine and Lunar Prospector missions that most likely indicate large quantities of water are frozen in cold traps at the lunar poles[1]. In 1996, the Clementine mission discovered permanently shadowed craters at both poles of the Moon -- the large Aitken basin in the south, and a series of smaller craters in the north. There may also exist permanent shadows in the bottoms of deep craters as much as 25 degrees from the poles. One preliminary radar experiment on Clementine postulated the existence of ice in these cold traps.

Preliminary data analysis from Lunar Prospector indicates that there are 260 million metric tons (MT) of ice at the lunar poles, with 200 million MT in the south and 60 million MT in the north. The data are not as conclusive in the north because the diameter of the cold trap craters there is near the resolution of Lunar Prospector's instruments[4].

To date, no lunar water-based ISRU operations have been attempted. Therefore, it is difficult to generate useful cost figures for this propellant production system. Orbital Technologies recently performed a lunar transportation architecture study to evaluate the effects of different levels of ISRU[5]. Their overall evaluation criterion was Earth launch mass (ELM). The architecture includes two reusable vehicles, an orbital transfer vehicle and a lander, and maintenance/propellant resupply depots in LEO, LLO, and on the lunar surface. The launch mass savings and ETO launch cost results of the study are shown in Table 2 and Table 3 respectively. Utilizing both lunar hydrogen and lunar oxygen leads to ELM savings of 67% in this case.

Table 2: Launch Mass Savings

	No ISRU	Lunar LOX	Lunar LOX & LH2
ELM	8000 MT	3900 MT	2600 MT
% Savings	-	52.50%	67.50%

Table 3: ETO Launch Cost

	No ISRU	Lunar LOX	Lunar LOX & LH_2
at $10,000/kg	$80 billion	$39 Billion	$26 Billion
at $1,600/kg	$12.8 Billion	$6.24 Billion	$4.16 Billion

System Scale and Cost

The major difference between available studies of ISRU facilities and the MARTA lunar transportation architecture is the scale of operation. In 1993, Sherwood and Woodcock sized an oxygen production facility to produce 100 MT of propellant per year[3]. Since one of Sherwood and Woodcock's landers required 25 MT of propellant to make one flight from the lunar surface to LLO and back, the production capability allowed them to make four such flights per year[3]. Production facility mass was 190 MT.

In the baseline MARTA architecture, with market growth of 5% per year, annual ISRU propellant production requirements ramp up from 1800 MT in year one to 4000 MT in the final year of the program 17 years later. Assuming 100% efficient extraction of the 2% of ice crystals in the cold trap regolith, a 30 MT batch of regolith yields 0.6 MT of water. Producing 2000 MT of propellant annually requires 3300 batches or 100,000 MT of processed regolith in a continuous process. In 1999, a graduate team at Caltech's Laboratory for Space Mission Design examined a facility for producing oxygen and hydrogen from lunar polar ice and generated the curve in Figure 4 for facility mass as a function of required annual propellant[6]. For reference, the Sherwood and Woodcock data point also is included on the figure. Their model of the cold trap regolith assumed water to be 14% by mass of the cold trap regolith; more recent analysis indicates there is only 2% by mass. Their plant mass to produce 2000 MT of propellant annually is 25 MT, much less than the 190 MT required in the Boeing study to produce just 100 MT of oxygen annually

Obviously, the different studies provide widely varying results. Such a wide distribution makes it hard to confidently input ISRU cost into a design model. As a result, ISRU cost was treated parametrically for the MARTA project.

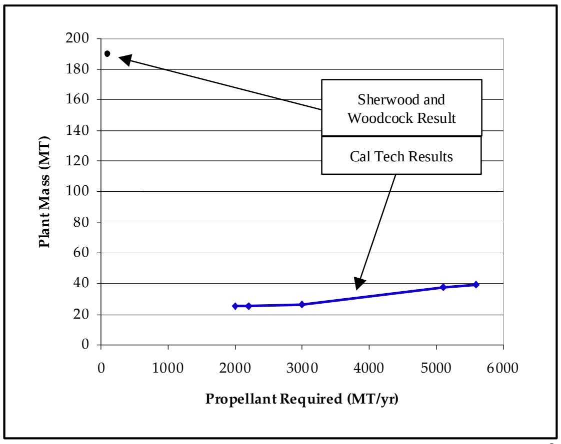

Figure 4: Production Facility Mass vs. Propellant Required[6]

Lunar Surface Architecture Selection Process

In order to make the chosen architecture work financially, the propellants needed to fuel the rocket vehicles must be produced on the lunar surface. Since substantial amounts of ice exist at the lunar poles, it makes sense to locate a propellant production facility at one of them. Because some of the missions will be to the equator, there needs to be a way to refuel the vehicles landing at the equatorial site. Options considered for moving propellant from the poles to the equator included various combinations of lander vehicles, roving trucks, and a mass driver. The landing vehicles would be used to land at either the equator or poles and have the capability to jump from base to base if needed. The roving truck would be capable of navigating the 2730 kilometers from the polar base to the equator allowing transfer of cargo, people and propellant. The mass driver would be used to launch propellant into LLO.

The mass and power requirement of the truck vehicle as well as the enormous travel distance required were deemed too difficult without excessive DDT&E costs. These technical and financial difficulties removed the truck from consideration. The remaining options were narrowed to the following choices: 1) a two-lander system with one vehicle sized for equatorial landings and the other for polar missions 2) a single lander that would land at both bases 3) a single lander in conjunction with a mass driver for launching propellants into LLO.

The required mass, propellant usage, and program cost for each option was calculated for the remaining candidates. Parametrically varying the ISRU propellant price per kilogram allowed the design team to generate the graph in Figure 5.

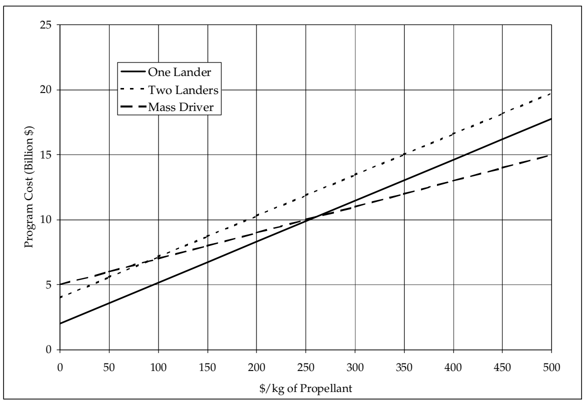

Figure 5: Lunar Surface and LLO Architecture Study

Perhaps the most valuable information obtained from Figure 5 is the fact that the single lander line intersects the mass driver line at $250 per kilogram. This implies a trade-off exists between the two configurations. If propellant can be made cheaply on the Moon, then it is best to use an all-lander system that uses a large amount of propellant. However, if propellant is very costly to produce on the lunar surface, the propellant savings of using the mass driver make this option more appealing. This emphasizes the fact that defining the final system configuration cannot be done unless ISRU cost is determined with confidence.

Simulated Probabilistic Parametric Lunar Architecture Tool Development

In order to calculate the mass, size, and cost of the transportation system being designed, it was necessary to create various models. These models needed to be flexible so that they could adapt to changes in the project as it was refined throughout the design process. The following sections detail the development of the Simulated Probabilistic Lunar Architecture Tool (SPPLAT).

Weights and Sizing

A traditional Weight Breakdown Statement (WBS) was used in the formulation of the Weights and Sizing (W&S) model.

This model minimizes the dry weight and propellant used for a given engine specific impulse (I_{sp}) and a combined Weight Adjustment Factor (WAF). This WAF was composed of two separate parts. The first was a Technology Reduction Factor (TRF) that modeled how much the dry weight could be reduced due to advances in materials technology. The second was a weight estimating error that modeled the inaccuracies in the W&S model itself. Both factors were expressed as percentages, and they were multiplied together to form the combined WAF.

Using Monte Carlo techniques, a Response Surface Equation (RSE) was generated from 110 converged point designs that spanned the design space. This RSE was then used as the W&S model in the overall design tool, SPPLAT.

Costing and Business Analysis

In order to determine the profitability of the business, a cost and business model was created to estimate the following values:

1) Costing of the Lunar Lander and Transfer Vehicles (LLTVs) using weight-based parametric Cost Estimating Relationships
2) Fleet size estimation and acquisition
3) Mass driver costing and payload capacity
4) Income and cash flows statements for calculation of project Net Present Value, (NPV)

The cost of the LLTVs was determined using weight-based Cost Estimating Relationships (CERs). The CERs used were from the 1996 NASA Air Force Cost Model (NAFCOM96)[7]. These CERs are based on shuttle-era launch vehicle technology, and in many ways do not reflect the actual nature or technology of an in-space vehicle. However, since no reusable in-space transfer vehicle has ever been constructed, there are currently no CERs directly applicable to this project. In order to account for the differences between the hardware represented in the NAFCOM96 CERs and MARTA's LLTVs, the CERs were multiplied by complexity factors to adjust the estimated cost up or down to obtain a more realistic cost model of the LLTV.

The LLTV costs were divided into two areas, DDT&E and a Theoretical First Unit cost (TFU). DDT&E represents all of the engineering and prototyping efforts required prior to the manufacture of the first vehicle. TFU represents the cost of building a single vehicle, with no learning curve or rate effects included. This analysis assumed that the main engine would be an off-the-shelf item, and that the RCS thrusters would be available off-the-shelf with only minor modifications. Most likely, this engine will be something similar to the RL60 engine under development by Pratt and Whitney. The RL60 is being designed to produce at least 50,000 lbf of vacuum thrust with a corresponding vacuum I_{sp} of approximately 460 sec. As a result, no DDT&E for main engines was included, and a substantially reduced DDT&E for RCS thrusters was used. The complexity factors used in the costing model are included in Table 4.

Table 4: LTV Complexity Factors

Vehicle Weight Group	DDT&E Complexity	TFU Complexity
Structure & Tank	0.8	1.0
RCS	0.1	1.0
Aerobrake	0.8	1.0
Primary Power	0.5	0.5
Electrical Conv/Dist	0.5	0.5
Environmental Contro	0.2	0.5
Avionics	0.2	0.7
Main Engine	0.0	1.0

As shown in Table 4, substantial reductions were assumed for primary power, electrical conversion/distribution, environmental control and avionics DDT&E and TFU. Since substantial technological changes have occurred in these areas since the Shuttle development, this was deemed appropriate. The other TFU costs were left unchanged in order to be conservative. In addition to these hardware-related costs, costs were included for various systems and testing operations. These were calculated as a percentage of total hardware costs. In addition to all of the above costs, a 20% margin was included to account for miscellaneous program costs that might be incurred.

Design of Experiments (DOE)

In order to gauge the effects of varying the ISRU cost on the overall economics of the project, a design of experiments (DOE) matrix was set up to perform a response surface analysis using SPPLAT. Because the use of a lunar mass driver was handled as a discrete variable, two separate response surfaces were created. Both response surfaces used ISRU cost, rocket engine I_{sp}, and weight adjustment factor (WAF) as control variables.

In order to make the design more robust, an uncertainty analysis using Monte Carlo simulation was also performed. The mass estimate, cost estimate, market expansion rate, and ETO cost per kg were allowed to vary between the limits shown in Table 5. The flow of this process is illustrated in Figure 6. For a given run of the DOE, 5000 Monte Carlo iterations were performed. For each iteration, a random value was picked within the range of each of the noise variables. The Monte Carlo analysis provided mean and standard deviation response surfaces. The end result was a group of response surface equations capable of modeling the output parameters over the entire range of the inputs for both architecture selections. The RSEs of interest in this project are:

1) Price to charge the customer that results in a 25% rate of return for the business,
2) The vehicle dry mass
3) Propellant required to complete on cargo transfer

A sample response surface is shown in Figure 7. For simplicity, this surface demonstrates the effect on vehicle dry weight of varying I_{sp} and ISRU cost. The color contours are used to help show the curvature of the surface. The optimal design was selected by using SPPLAT to find the combination of control variables that resulted in the minimum price to charge the customer. The uncertainty analysis using the noise variables allowed the design team to associate a confidence level with this price to charge. In other words, the uncertainty analysis allows the design team to assess how likely it is that a combination of control variables will minimize the price to charge.

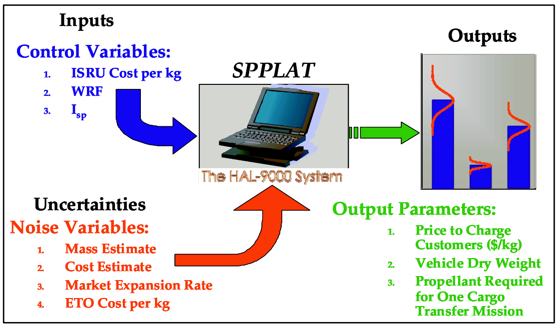

Figure 6: Uncertainty Analysis Flowchart

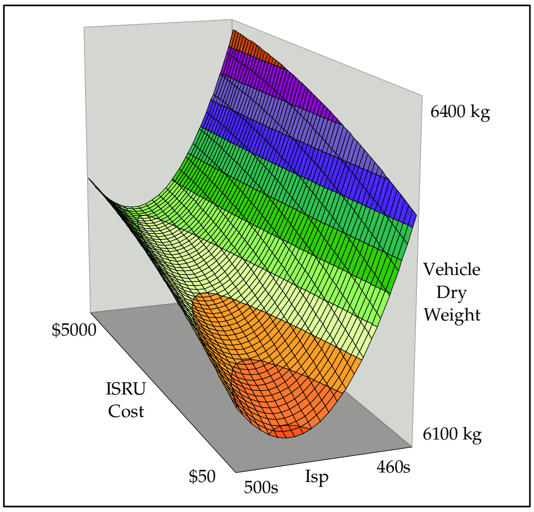

Figure 7: Representative Response Surface

Table 5: Noise Variable Ranges for the Monte Carlo Simulation

Noise Variable	Minimum	Most Likely	Maximum
Mass Estimate	-20%	0%	25%
Cost Estimate	-5%	5%	15%
Market Expansion	0%	5%	15%
ETO Cost per kg	$800	$1,600	$5,000

Results

This section contains the details related to the major decisions made during the design process.

Trajectory Description

An example cargo transfer scenario starts on the Moon's surface at the South Pole as shown in Figure 8. Two vehicles are required for the entire mission, the first carrying the cargo and the second carrying additional propellant for refueling. The cargo vehicle leaves the Moon's surface carrying 30 MT of cargo and 109.7 MT of additional propellant. The refueling vehicle carries 139.7 MT of additional propellant. Both vehicles burn 46.6 MT of propellant to produce the 1700 m/s ΔV necessary to reach LLO.

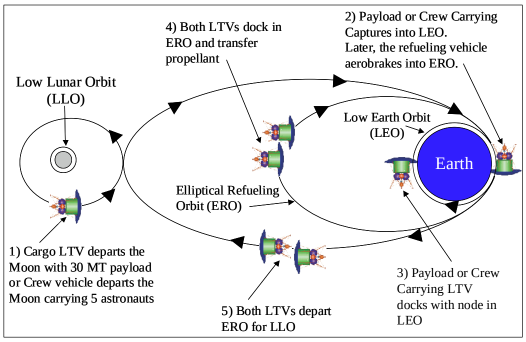

Figure 8: Sample Transfer Scenario

Once in LLO, each vehicle takes 16.4 MT from its additional propellant in order to make the 800 m/s ΔV for the TEI burn. Both vehicles then spend 5 days in transit to Earth. The cargo vehicle conducts 12 aerobrake passes (adding another 5 days to the transfer) through the atmosphere to produce the ΔV of 3100 m/s needed to capture into LEO. It then performs a rendezvous with the transportation node and swaps out the 30 MT cargo for 60 MT of outbound cargo. Not needing to be in LEO, the refueling vehicle aerobrakes directly into the ERO where it will rendezvous with the cargo vehicle

Once the cargo vehicle has completed the cargo transfer and any necessary maintenance, it uses all its remaining propellant to make the 2400 m/s ΔV needed to enter an ERO where it will meet the refueling vehicle to take on the propellants needed to get back to the Moon. At this point, the cargo vehicle takes on sufficient propellant to complete the trip to the Moon leaving enough propellant for the refueling vehicle to make the same trip.

Because the cargo mission outlined above takes too much time to comfortably transfer astronauts using the same methods, a separate mission scenario was developed for astronaut missions. The main difference between the two scenarios is found in the leg of the trip from the Moon to LEO. Instead of the aerobraking procedure used with the cargo, the astronaut missions use the MARTA vehicle rocket engine to provide the ΔV necessary to capture into LEO. This maneuver is possible because the crew module is small enough that the vehicle can carry enough propellant to successfully complete the maneuver. Once the vehicle carrying the astronauts leaves LEO, it follows the same procedure as the cargo mission.

Mass Driver Description

Various mass driver designs were considered in an attempt to find the best one for the mission. The mass driver chosen for this project operates by accelerating the payload using magnetic attraction. The magnetic field is generated by a linear synchronous motor timed by feedback of the payload's position along the track. The final section of the track is devoted to dampening any disturbances and correctly aligning the payload to minimize trajectory error. The payload will have some reaction control correction ability to correct for any small launch spread. The chosen system is powered by nuclear generators although solar power could be used if political considerations make use of nuclear power an issue. An efficiency of 92% is assumed for the conversion of electrical energy to kinetic energy.

The mass driver system breakdown is provided in Table 6 below. All mass, power and cost estimates are based on relationships found in reference 8. The baseline design is sized to generate the ΔV of 1700 m/s that is required for LLO insertion. The 20 Earth-g load requirement was found to be a good compromise between excessive track length and the maximum loading the structural system could reasonably handle. The mass of propellants launched per year is calculated from the number of cargo flights multiplied by their propellant usage requirement. The “chunk” size represents the mass of the payload launched by each shot of the mass driver. It was determined that 30 MT would be most convenient if the mass driver is to be used later for launching cargo.

Table 6: Baseline Mass Driver System Requirements

ΔV to Reach LLO	1,700 m/s
Mass Launched per Year	2,000,000 kg/yr
Number of g's at Launch	20
"Chunk" Size	30,000 kg
Length of Track	7,400 m
Total Launcher Mass	36,800 kg
Total System Mass	57,600 kg
Total Power	295,000 W
Estimated Annual Recurring Cost	$919,300
Estimated Non-Recurring Cost	$1,922,900,000

Baseline Cost Breakdown

A profitable 25% rate of return was set in the business case, and cost per kilogram of lunar propellants was varied, along with engine I_{sp} and weight technology reduction factor. ISRU cost was the driving parameter, followed by use of a mass driver. Customer price is fairly insensitive to engine Isp and WAF. Varying lunar propellant cost leads to variation in the price charged to the customer for transporting cargo from LEO to the lunar surface. The results of the team's trade study are shown in Figure 9.

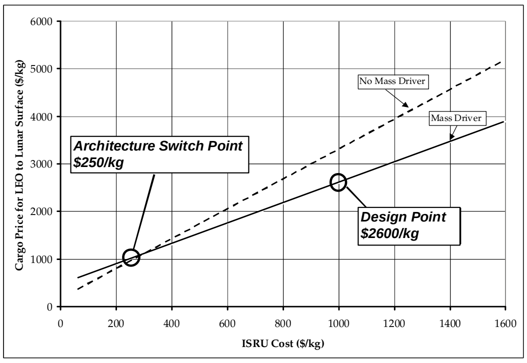

Figure 9: Customer Price as a Function of ISRU Cost

If propellant price can be brought down to $160/kg, the original RFP price goal of $800/kg can be achieved. The team feels that a propellant price of $1000/kg, which yields a cargo price of $2600/kg, is a reasonable goal that can motivate ISRU technology development over the next 18 years before IOC.

Using SPPLAT’s cost model, a cost breakdown was found for the baseline vehicle as shown in Table 7. The price to charge customers per kg for transfer from LEO to the Moon was the main output of the model based on obtaining an NPV of zero with a discount rate of 25%. The largest expense was approximately $48 billion for ISRU propellants over the life of the program.

Table 7: Baseline Cost Breakdown

Price to Charge Customers for LEO to Moon Transfer	2600/kg
IRR	25%
NPV	$0
Vehicle DDT&E	$1,000 M
LTV ETO Launch Costs	$570 M
ISRU Propellant Costs	$47,650 M
Mass Driver DDT&E	$2,300 M
Operations Costs	$1,500 M
Fleet Acquisition Costs	$1,000 M
Life Cycle Costs	$54,000 M
Total Revenue	$74,000 M

Results of the Design of Experiments

The results of the DOE provide a robust assessment of the effects of the control variables, also showing the effects of uncertainty in the design relationships via the noise variables. The RSEs themselves are very accurate. Goodness of fit analysis shows that the equations possess very high R-squared (R^2) values. High R^2 values indicate a good match between the RSE and the original data points. With the exception of the vehicle dry mass standard deviation equation, all of the R^2 values are above 0.996.

The RSE's show that the price to charge the customer per kilogram of payload should be set to $2600/kg of cargo and $2 million/person to provide a 25% rate of return for the baseline design. These price figures require the use of a lunar mass driver because the baseline ISRU cost is high enough to warrant its use. If the design is implemented without the use of the mass driver, the prices to charge the customer increase by approximately 22%. Using the available standard deviation RSE's, the optimum price combination shows that the price will fall within 7% of the quoted mean prices with 95% confidence levels.

Because the number of astronaut flights is smaller than the number of cargo flights, the price to charge per astronaut does not change noticeably. For cargo missions, within the range of input variables specified, the minimum possible price to charge is $307/kg. This price results when a lunar mass driver is not used, the engine I_{sp} is increased to 500 seconds, the cost per kilogram for ISRU production is brought to $50, and a 20% technology reduction factor (TRF) used.

A comparison between the baseline vehicle and different designs is shown in Figure 10. Increasing the I_{sp} of the rocket engine to 500 sec only reduces the price to charge the customer for a kilogram of cargo to $2373/kg, and increasing the TRF to 20% only reduces the price to $2498/kg. The combined benefit of implementing both advances in technology provides a savings of 12% to the customer. However, investing in ISRU technology and reducing the cost per kilogram of ISRU production to $50 results in a savings of 86%. It should be noted that the use of a lunar mass driver is no longer beneficial once the cost of ISRU propellants is brought below $250/kg. Therefore, the cost of ISRU propellants has a significant impact on the economics of this design. Not only does a low ISRU cost allow the price per kilogram of payload to reach very low levels, but it also removes the need to invest in additional technology, namely the lunar mass driver. Figure 11 shows how sensitive the price to charge the customer is to the cost of ISRU propellant production.

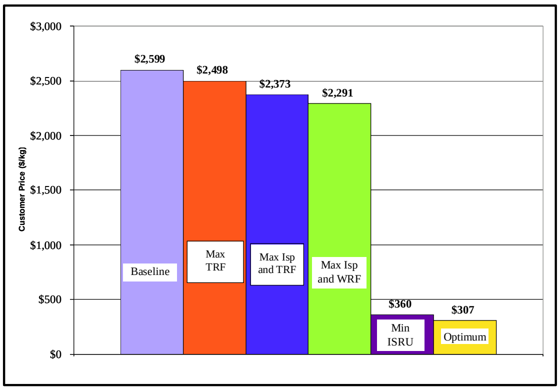

Figure 10: Price to charge customer per kg of payload for the optimal and baseline design cases

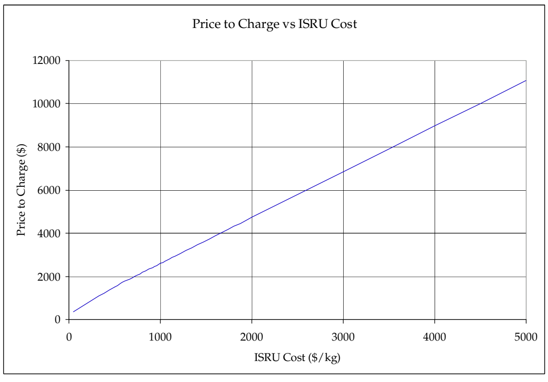

Figure 11: Effects of ISRU Cost on the Price to Charge the Customer

Conclusions and Recommendations

The main conclusion reached from this project is that it is currently possible to build a commercially viable and technologically feasible Earth-Moon transportation system even though it would be costly. The MARTA vehicle presented does not rely on any advanced technologies or require any technical advances to become a reality. However, the most important feature of the architecture is not the vehicle. In order to make this a profitable venture, the cost of producing propellants on the Moon must be controlled. In fact, this one technology is the single largest factor in determining how much a company must charge in order to make a 25% return. As such, NASA or other similar groups should focus resources on developing a low cost lunar ISRU facility.

Another important result of the study is that the use of a mass driver is not a necessary requirement for the system as outlined. In fact, it only improves the business case for the system when the cost of ISRU production is in excess of $250/kg. This fact reiterates the importance of lowering the cost of an ISRU facility. By reducing the cost below $250/kg, it is possible to significantly reduce the complexity of the system and time needed to develop and deploy it because the mass driver is no longer necessary.

The final conclusion is that moderately improving the I_{sp} of liquid oxygen/liquid hydrogen fueled rocket and reducing the mass of the vehicle through advanced materials technologies does help reduce the cost of the system. But, the effects are only marginal. As a result, the MARTA team does not feel it is justified to spend research dollars trying to improve these two technologies when today's technologies work almost equally as well. Instead, all resources should be concentrated on lowering the cost of an ISRU facility.

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