Moon-based Advanced Reusable Transportation Architecture
The *MARTA* Project

Georgia Institute of Technology
Graduate Program

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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. This investigation was conducted from the perspective of a private company operating the transportation system for a profit. The goal of this company was to provide an Internal Rate of Return (IRR) of 25% to its shareholders.

The technical aspect of the study began with a wide open design space that included nuclear rockets and tether systems as possible propulsion systems. Based on technical, political, and business considerations, the architecture was quickly narrowed down to a traditional chemical rocket using liquid oxygen and liquid hydrogen. However, three additional technologies were identified for further investigation: aerobraking, in-situ resource utilization (ISRU), and a mass driver on the lunar surface.

These three technologies were identified because they reduce the mass of propellant used. Operational costs are the largest expense with propellant cost the largest contributor. ISRU, the production of materials using resources on the Moon, was considered because an Earth to Orbit (ETO) launch cost of $1600 per kilogram made taking propellant from the Earth's surface an expensive proposition. The use of an aerobrake to circularize the orbit of a vehicle coming from the Moon towards Earth eliminated 3,100 meters per second of velocity change (Delta V), eliminating almost 30% of the 11,200 m/s required for one complete round trip. The use of a mass driver on the lunar surface, in conjunction with an ISRU production facility, would reduce the amount of propellant required by eliminating using propellant to take additional propellant from the lunar surface to Low Lunar Orbit (LLO). However, developing and operating such a system required further study to identify if it was cost effective.

The vehicle was modeled using the Simulated Probabilistic Parametric Lunar Architecture Tool (SPPLAT), which incorporated the disciplines of Weights and Sizing, Trajectories, and Cost. This tool used ISRU propellant cost, Technology Reduction Factor (a dry weight reduction due to improved technology), and vehicle engine specific impulse as inputs. Outputs were vehicle dry weight, total propellant used per trip, and cost to charge the customer in order to guarantee an IRR of 25%. SPPLAT also incorporated cost estimation error, weight estimation error, market growth, and ETO launch cost as uncertainty variables. Employing SPPLAT over a range of inputs produced the following results.

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. The best estimate of ISRU cost is $1000/kg, and results in a price to charge the customer of $2600/kg of payload. 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 was only cost effective at an ISRU propellant cost greater than $250/kg, although it reduced total propellant used by 35%.

In conclusion, this mission is achievable with current technology, but is only profitable with greater research into the enabling technology of ISRU propellant production.

Acronyms

AHP	Analytic Hierarchy Process	MARTA	Moon-based Advanced Reusable Transportation Architecture
CER	Cost Estimating Relationship	NAFCOM96	1996 NASA Air Force Cost Model
DDT&E	Design, Development, Testing, and Evaluation	NPV	Net Present Value
EBIT	Earnings Before Interest and Taxes	RFP	Request For Proposals
ELM	Earth Launch Mass	RSE	Response Surface Equation
EOI	Earth Orbit Insertion	RSM	Response Surface Methodology
ERO	Elliptical Refueling Orbit	SPPLAT	Simulated Probabilistic Parametric Lunar Architecture Tool
ETO	Earth To Orbit	TEI	Trans-Earth Injection
GEO	Geostationary Orbit	TFU	Theoretical First Unit cost
IRR	Internal Rate of Return	TLI	Translunar Injection
ISRU	In-situ Resource Utilization	TRF	Technology Reduction Factor
LEO	Low Earth Orbit	WBS	Weight Breakdown Statement
LLO	Low Lunar Orbit	WAF	Weight Adjustment Factor
LLTV	Lunar Lander and Transfer Vehicle		
LTV	Lunar Transfer Vehicle		

1. 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. 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 Artemis Society International are advocating the establishment of privately financed permanent human colonies on the Moon for the sole purpose of making a profit.

While seemingly unrelated at first glance, 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 goals of the project were 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 envisioned 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 include that all of the astronauts taken to the Moon must be returned to LEO, while 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%. 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 2018, the initial year of operation. A final requirement for the design to be successful 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.

2. Problem Approach

The MARTA team took a novel approach to the design process. In an attempt to provide oversight and reduce mistakes, the whole team was divided into two smaller teams, the Design Team and the Review Team. The Design Team went through the steps outlined in the sections that follow and periodically provided the Review Team with data. The Review Team then performed their own completely independent analysis to verify or refute the results generated by the Design Team. If the two results differed, the Review Team would offer suggestions and generate "what if" scenarios to insure that the Design Team considered all of the possibilities.

2.1 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 expected 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 (See Section 2.2 for more details). 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.

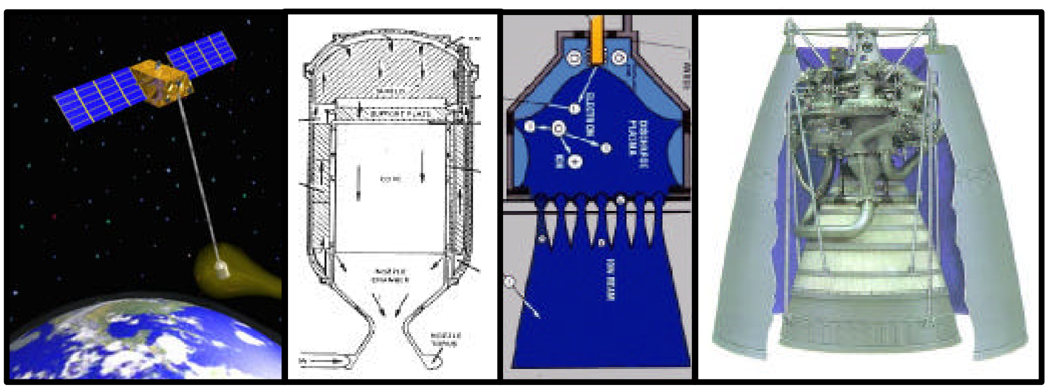


Figure 1: Propulsion Systems Considered

With these four systems identified, more detailed analyses provided a more complete idea of the main benefits each offered as well as the main drawbacks to the systems. The detailed analysis also allowed for a systematic down-selection process that resulted in a single architecture. To ensure an unbiased down-selection process, the design team employed a tool called the Analytic Hierarchy Process (AHP). AHP allowed direct comparison of each candidate architecture to each of the other architectures on a one-to-one basis. The process highlighted the strengths and weaknesses of each candidate and allowed the team to pick the overall strongest option. The results of the AHP 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. The third candidate, an electric propulsion system, was eliminated because of time considerations. The current state of the art in electric propulsion required a three-month period to move a satellite from LEO to Geostationary Orbit (GEO). As such, it would take too long to move a vehicle from LEO all the way to the Moon. This left the chemical liquid rocket system that used 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. For safety reasons, the team decided against employing the aerobraking procedure on the astronaut transfer missions.

An additional method of reducing overall propellant use was the implementation of numerous fuel depots, including one in 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.

In order to maintain the usefulness of in-space refueling, a just-in-time refueling plan was developed. Using additional vehicles to carry the additional propellant needed, the orbital precession of a fuel depot was avoided, as the refueling vehicle would be sent only as needed.

2.2 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.

2.2.1 Economics of Lunar Propellants

The team decided to investigate the use of lunar propellants in its lunar transportation architecture for two reasons. First, 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 propellants 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 [1]. The baseline architecture in this study has a payload/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.

2.2.2 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 [2]. 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.

Two billion years ago, the Moon was close enough to the earth that its axis of revolution was unstable and there were no cold traps on the lunar surface. As the Moon's distance from Earth increased, its axis stabilized and ice from comet and meteor impacts began to accumulate in permanent shadow. Constant bombardment by meteors led to mixing of the ice deposits with surrounding regolith and prevented its dispersal by sublimation. About two meters of regolith has accumulated in this fashion since the formation of the cold traps, so ice is not expected below that depth [2].

Lunar Prospector's neutron spectrometer measured the flux of neutrons of various energies scattering off of the hydrogen trapped in the surface regolith. Figure 2 shows maps of hydrogen concentration at the lunar poles based on these measurements [3]. The darker color represents larger amounts of hydrogen, which indicates the presence of water. Preliminary data analysis 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 less sure in the north because the diameter of the cold trap craters there is near the resolution of the neutron spectrometer. Better results will become available in late 2000 after further data reduction [4].

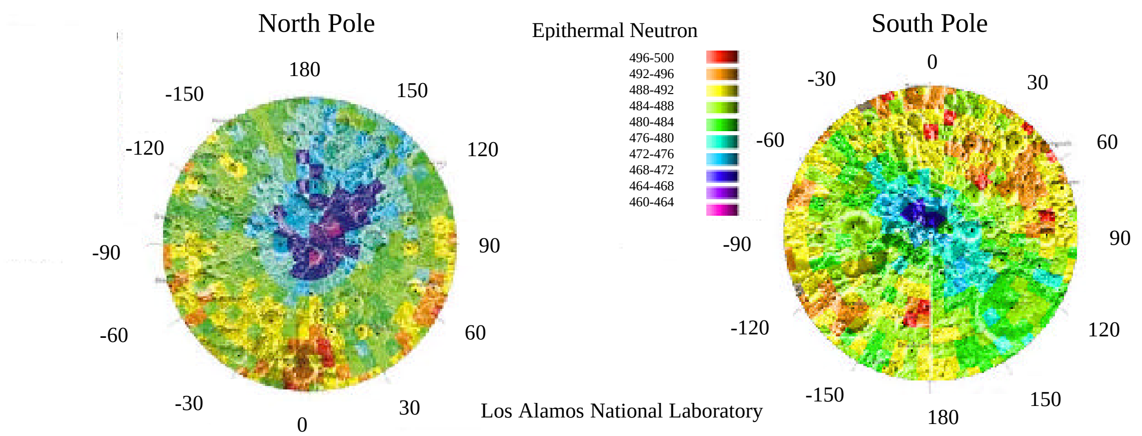


Figure 2: Polar H2 Concentrations from Lunar Prospector Neutron Spectrometer [3]

Producing liquid hydrogen and oxygen propellants from lunar polar ice involves several functional groups:

1) Autonomous rovers for regolith feed/slag transportation
2) Solar thermal furnace for water evaporation
3) Condenser for water vapor collection
4) Electrolysis unit for production of oxygen and hydrogen from water
5) Heat exchanger for liquefaction of propellants
6) Cryogenic storage system

The rovers must work in the extreme conditions of permanent shadow, and the scale of the operation could tax rover automation or strain its mass budget. The solar thermal furnace should be simple enough, given its location on a crater rim in permanent sunlight and the low temperatures required for evaporation compared to other ISRU techniques to be described. Water electrolysis is a space-proven system in the Russian Mir space station's Elektron oxygen generation unit, and in reverse in the space shuttle's fuel cells. Finally, cryogenic storage in the cold traps should be simple. Thus, it appears that the main technical challenges confronting the development of such a system are related to collection and dispersal of the regolith due to the cold operating temperatures and high material throughput.

Given these uncertainties, it is difficult to generate useful cost figures for this propellant production system. Orbital Technologies of Madison, WI recently performed a lunar transportation architecture study to evaluate the effects of different levels of ISRU [3]. 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. Nominal mission length for this study is twenty years. The launch mass savings and ETO launch cost results of the study are shown in Table 1 and Table 2 respectively. Utilizing both lunar hydrogen and lunar oxygen leads to ELM savings of 67% in this case. Before trying to quantify this result in a cost model, it will be helpful to look at other ISRU techniques that have been researched other groups.

Table 1: Launch Mass Savings

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

Table 2: ETO Launch Cost

	No ISRU	Lunar LOX	Lunar LOX & LH2
at $10,000/kg	$80 billion	$39 billion	$26 billion
at $1,600/kg	$12.8 billion	$6.24 billion	$4.16 billion

2.2.3 Other ISRU Methods

Prior to the discovery of ice at the lunar poles, ISRU research focused on the production of oxygen from regolith. Oxygen composes an average of 40% of lunar regolith. There are three main methods of extraction: chemical reaction, vacuum pyrolysis, and silica melt electrolysis [5]. According to Lunar Prospector's principal investigator, Dr. Alan Binder, no detailed research has yet been done on evaporation and electrolysis of polar ice. As a result, the closest reference process would be vacuum pyrolysis. Both processes involve simple heating of lunar material, but vacuum pyrolysis of dry regolith requires much higher temperatures, on the order of 2000 K, before useful products result. Vaporizing water from cold-trap regolith would require heating only to 400 K, just above the boiling point of water. Vacuum pyrolysis techniques need not deal with the cryogenic temperatures faced in cold traps, but since the process would probably occur away from the poles, the facility would either stand-down half the time or incur a mass penalty due to a power storage system for operation during the lunar night. Current state of the art vacuum pyrolysis, used widely in earth-based metal processing, uses batch sizes of 30 MT [5].

2.2.4 System Scale and Cost

The major difference between available studies of pyrolysis 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. 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 [1]. 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 3 for facility mass as a function of required annual propellant [6]. For reference, the Sherwood and Woodcock data point is also 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. Due to the widely varying results of current studies, ISRU cost was treated parametrically for the MARTA project.

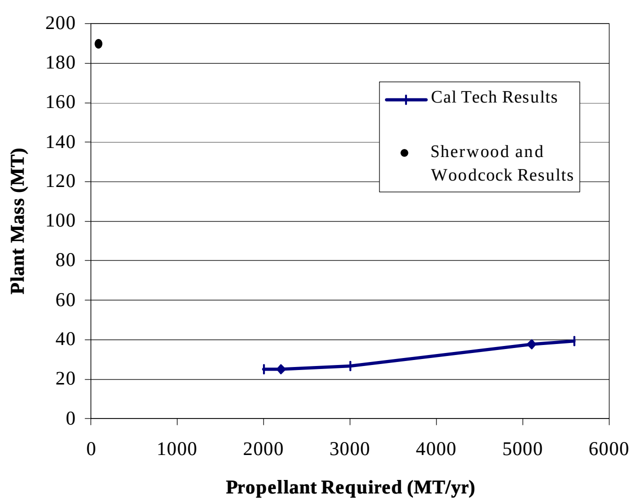


Figure 3: Production Facility Mass vs. Propellant Required [6]

2.3 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 the poles (See Section 2.2 for more details on this.). 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. This problem lead to an investigation intended to identify the optimal system architecture for transfer of propellant from the poles to the equator. Options considered 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 Low Lunar Orbit (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 4. Immediately evident is that the two-lander scheme has an overall higher program cost than a single vehicle option.

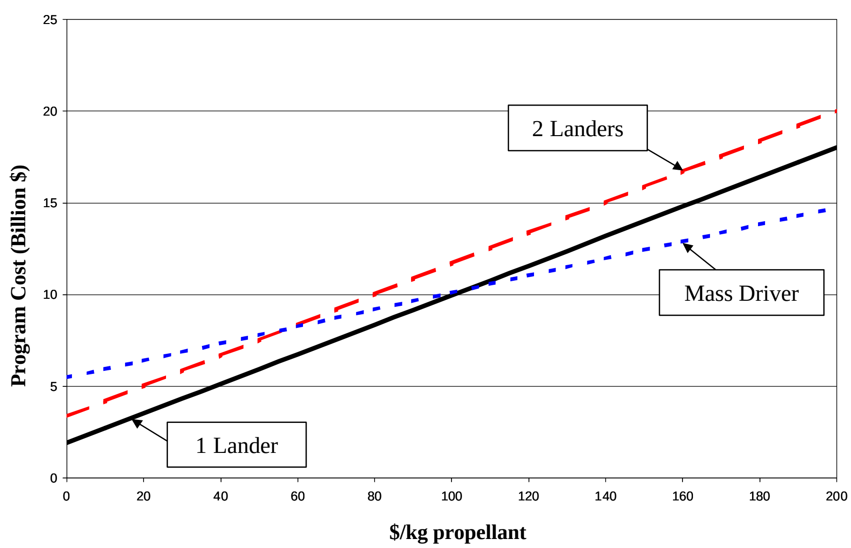


Figure 4: Lunar Surface and LLO Architecture Study

Perhaps the most valuable information obtained from Figure 4 is the fact that the single lander line intersects the mass driver line at $100 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 a more propellant hungry all-lander system. 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. The $100 per kilogram intersection was identified during this simplified trade study and does not reflect the final results. After more detailed analysis, the actual intersection was found to be at $250 per kilogram. As such, Figure 4 is included to underline the importance of ISRU cost to the system architecture. It also points out that defining the final system configuration cannot be done unless ISRU cost is determined with confidence.

2.4 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 how the Simulated Probabilistic Lunar Architecture Tool (SPPLAT) was developed.

2.4.1 Weights and Sizing

A traditional Weight Breakdown Statement (WBS) was used in the formulation of the Weights and Sizing (W&S) model. The Weight Breakdown Statement is provided in detail in Section 3.1.1.

This model used Solver, the Excel optimization routine, to minimize 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 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.

Response Surface Analysis was used in modeling the W&S for use in a Monte Carlo Simulation. Response surface analysis generates an equation for the desired variable, (e.g. dry mass of the vehicle) using the control variables as inputs. A Response Surface Equation (RSE) was generated from 110 converged point designs that spanned the design space. The control variables for the RSE were I_{sp} and the WAF. I_{sp} was varied from 450 seconds to 500 seconds in 5-second increments, while the WAF was varied from 80% to 125% in 5% increments. This RSE was then used as the W&S model in the ultimate design tool, SPPLAT.

In order to have this tool generate values for each line item of the WBS, it was necessary to be able to calculate component masses from the vehicle dry mass. The extreme cases of the design space were analyzed and line items were identified as either fixed or variable masses (For example, avionics were a fixed mass for this mission architecture that stayed constant while tank mass changes based on engine I_{sp}). The variable mass line items were proportioned to the dry weight remaining after the fixed mass items were removed. These ratios were then applied to the RSE value of the vehicle dry weight to calculate the line item masses. The reason for developing the tool in this manner was to allow SPPLAT to generate the entire WBS from a single RSE.

2.4.2 Costing and Business Analysis

In order to determine the profitability of the business, an Excel spreadsheet model was created that included the following functions:

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). These CERs are based upon 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, complexity factors were used to modify the cost by linear multiplication. The costs obtained from the CERs were multiplied by these 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 SPW2000 engine under joint development by Snecma and Pratt and Whitney. The SPW2000 is being designed to produce 50,000 lbf of thrust with an I_{sp} of 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 3.

Table 3: 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 can be seen, 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 percent of total hardware costs. The percentages used are shown in Table 4. In addition to all of the above costs, a 20% margin was included to account for miscellaneous program costs that might be incurred.

Table 4: LTV Non-Hardware Cost Percentages

	Complexity Factor Adjustment on NAFCOM Results for	Complexity Factor Adjustment on NAFCOM
System Test Hardware	20%	-
Integration, Assembly, Check	12%	25%
System Test Ops	14%	-
Ground Support Equip.	6%	-
System Eng. & Integration	20%	4.50%
Program Management	5%	5%

The fleet size is based on the number of round trips as well as necessary support flights needed each year. For each of these flight types, a total flight time, including ground processing and maintenance, was determined. Using these times, the total required fleet size was calculated for each year. The trip time and flight assumptions used are shown in Table 5. The assumption was made that any operations on the lunar surface (loading/unloading) require two days. Any flight that arrives in LEO will spend seven days there for maintenance and inspection. The vehicles in the fleet will be rotated through the different flight types so that all vehicles receive periodic maintenance in LEO.

Table 5: Trip Time and Flight Assumptions

Flight Type	Total Round Trip Flight Time (days)	Flights per Round Trip Cargo Flight	Flights per Round Trip Passenger Flight
Cargo (60 MT)	19	1.0	-
Passenger (5 people)	15	-	1
LLO Refueling	4	2	-
ERO Refueling	12	1	-
Equatorial Base Refueling	4	0.5	0.5

In any year that a larger fleet size is required than the previous year, the program is charged for the acquisition of a new vehicle. A learning curve effect of 95% was used for this acquisition. In other words, every time the total number of vehicles built doubles, the cost to acquire the next vehicle decreases by 5%. As shown in Table 6, this process resulted in maximum fleet sizes of 3 vehicles in the 0% and 5% growth cases. For the 15% growth case, the fleet size reaches 10 in the final program year.

Annual Flight Growth Rate	Max Number of Vehicles Required
0%	3
5%	3
15%	10

Table 6: Vehicles Required for Different Annual Growth Rates

Income and cash flow statements were prepared in order to calculate the project Net Present Value (NPV). A number of assumptions were made in the preparation of these statements. These assumptions are shown in Table 7 below.

Table 7: Accounting Assumptions Used for Income Statements

Fleet DDT&E Period	3 years
DDT&E Start Period	2014
Maintenance Costs per Cargo or Passenger Flight	$1 M
Lunar Surface Operations Cost	$10 M / yr
LTV Depreciation Method	Straight-line, 10 year lifetime
Mass Driver Depreciation Method	Straight-line, 15 year lifetime
Main Engine Life	100 flights

The cost of lunar propellants was treated as an independent variable, and the cargo revenue per kilogram necessary to produce zero NPV for a given cost of propellants was calculated. This step was necessary in order design a system that would meet the goal of returning a 25% rate of return for the private company that operates it. The goal was to calculate the price per kilogram that the company would need to charge NASA (the customer) in order to make the required return. This was easily accomplished by assuming that the market demand (the government-sponsored payload) would not change with changing cargo revenues. The added NPV generated by an additional $1 per kilogram of payload was determined. Since NPV is a linear operator, all that was required to determine the zero NPV cargo price was to divide the NPV for the nominal case by this $1/kg NPV. The result was then added or subtracted from the nominal cargo price to determine the zero NPV cargo price. This became the major financial metric used to evaluate the various mission configurations.

2.5 Setting up the Design of Experiments (DOE)

In order to gauge the effects of changing ISRU cost on the economics of the project, a design of experiment (DOE) matrix was set up to perform a response surface analysis using SPPLAT. Response surface analysis generates an equation for the desired variable, (e.g. price to charge customer) using the control variables as inputs. 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. The inputs for the design of experiments analysis are shown in Table 8.

Table 8: Design of Experiments Matrix

Run	x_1 Isp (sec.)	x_2 ISRU ($/kg)	x_3 Weight Red. (%)	Mass Driver
1	460	50	0	Yes
2	460	50	20	
3	460	5000	0	
4	460	5000	20	Yes
5	500	50	0	
6	500	50	20	
7	500	5000	0	Yes
8	500	5000	20	
9	480	2525	10	
10	460	2525	10	Yes
11	500	2525	10	
12	480	50	10	
13	480	5000	10	Yes
14	480	2525	0	
15	480	2525	20	
16	460	50	0	No
17	460	50	20	
18	460	5000	0	
19	460	5000	20	No
20	500	50	0	
21	500	50	20	
22	500	5000	0	No
23	500	5000	20	
24	480	2525	10	
25	460	2525	10	No
26	500	2525	10	
27	480	50	10	
28	480	5000	10	No
29	480	2525	0	
30	480	2525	20	

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 9. The flow of this process is illustrated in Figure 5. 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 (RSE) 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, and 3) Propellant required to complete on cargo transfer. A sample response surface is shown in Figure 6. 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 to team to associate a confidence level with this price to charge. In other words, the uncertainty analysis allows the design team to asses how likely it is that a combination of control variables will minimize the price to charge.

Table 9: 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

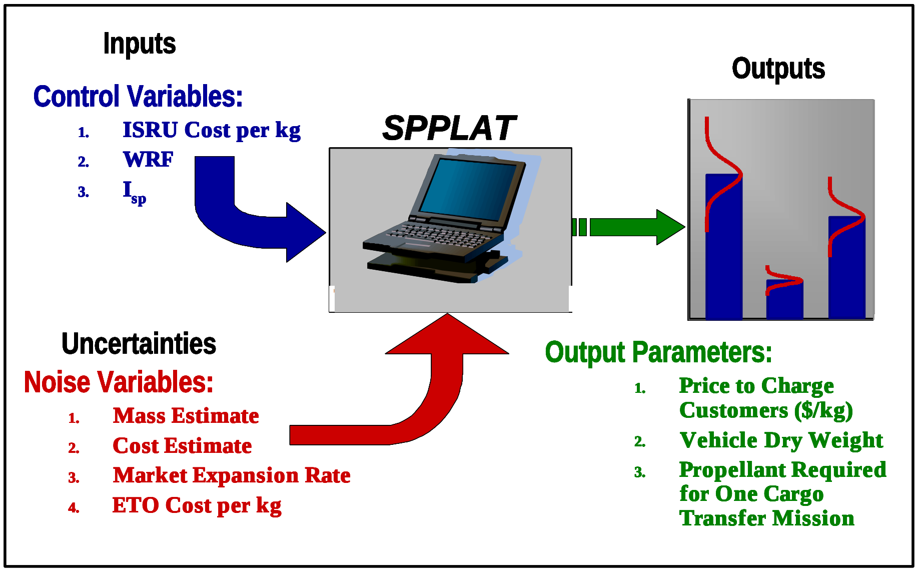


Figure 5: Uncertainty Analysis Flowchart

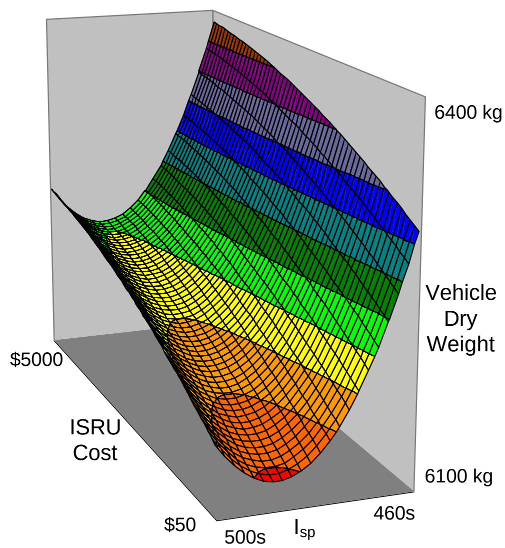


Figure 6: Representative Response Surface

3. Results

3.1 Baseline Operations/Architecture

The final 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. 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 to LEO and then inbound cargo or astronauts to the Moon's 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.

3.1.1 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, whereas a propulsive burn is used to capture a vehicle carrying astronauts. 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 7.

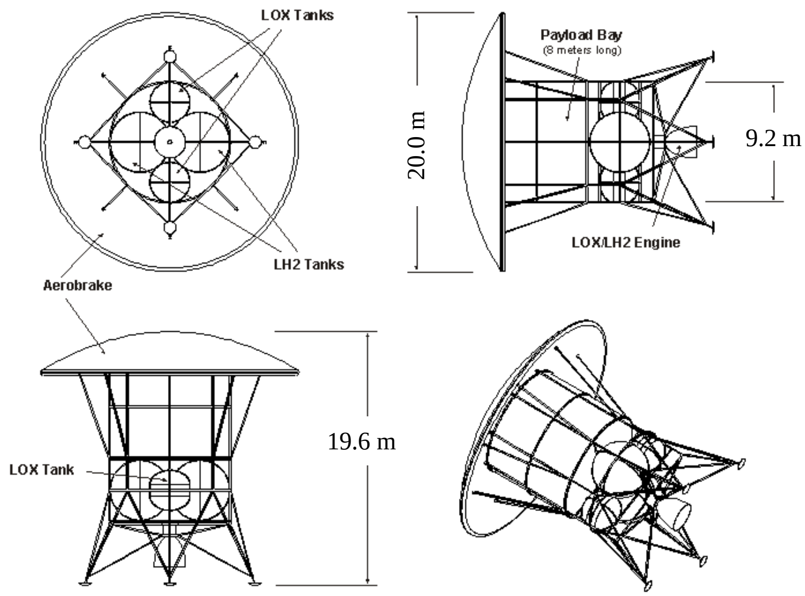


Figure 7: Three View of the MARTA Transfer Vehicle

The vehicle is designed to accommodate four different configurations, as shown in Figure 8. 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.

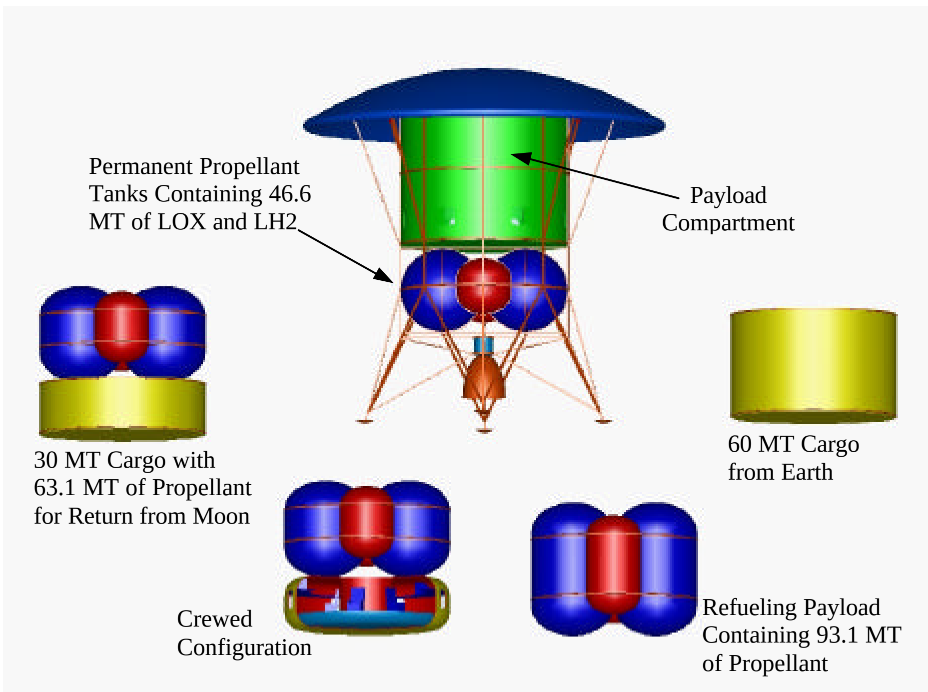


Figure 8: MARTA Transfer Vehicle with Various Configurations

Due to the low forces required for an in-space system, the vehicle itself is relatively light, as can be seen in the component weight breakdown given Table 10. The numbers in Table 10 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 10: 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**

3.1.2 Trajectory Description

An example cargo transfer scenario starts on the Moon's surface at the South Pole as shown in Figure 9. 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 returning cargo and 109.7 MT of additional propellant. The refueling vehicle carries 139.7 MT of additional propellant. Both vehicles burn all 46.6 MT of propellant in the permanently attached tanks in order to produce the 1700 m/s ΔV necessary to reach LLO.

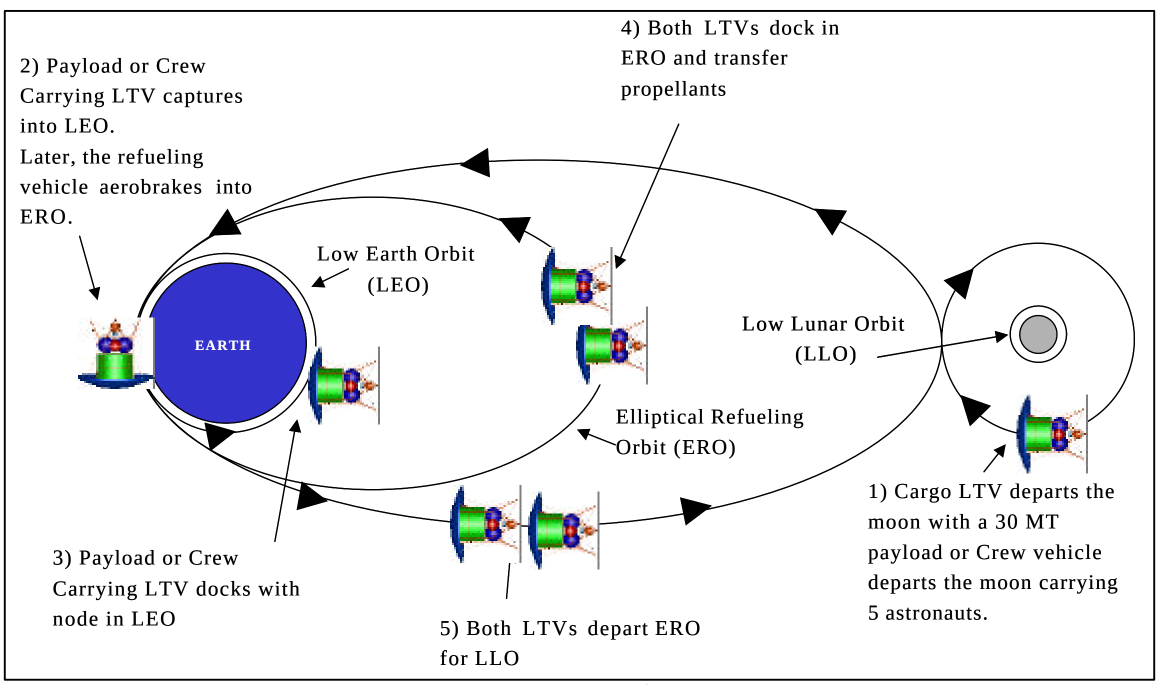


Figure 9: Sample Transfer Scenario

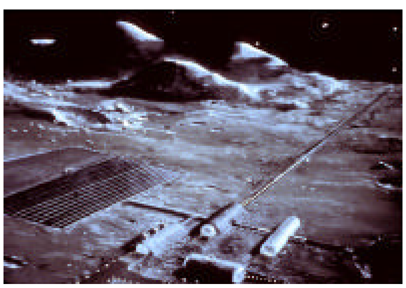
Once in LLO, each vehicle takes 16.4 MT from its additional propellant in order to make the 800 m/s ΔV for the TEI burn. Both vehicles then spend 5 days in transit to Earth. The cargo vehicle conducts 12 aerobrake passes (adding another 5 days to the transfer) through the atmosphere to produce the ΔV of 3100 m/s needed to capture into LEO. It then performs a rendezvous with the transportation node and swaps out the 30 MT returning cargo for 60 MT of outbound cargo.

Once the cargo vehicle has completed the cargo transfer and any necessary maintenance, it uses all its remaining fuel to make the 2400 m/s ΔV needed to enter an elliptical refueling orbit (ERO) where it will meet the refueling vehicle to take on the propellants needed to get back to the Moon. Not needing to be in LEO, the refueling vehicle aerobrakes directly into the ERO to rendezvous with the cargo vehicle. At this point, the cargo vehicle takes on sufficient propellant to complete the trip to the Moon. The cargo and refueling vehicles both burn to produce the 700 m/s needed to enter the transfer back to lunar orbit. At the Moon, each vehicle burns its engine to generate the 800 m/s of ΔV required to enter a polar LLO and then burns again for 1700 m/s to return to the surface. The cargo vehicle lands at either the South Pole or the equator (as required by cargo manifesting) while the refueling vehicle lands at the South Pole to begin the cycle again.

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.

3.1.3 Mass Driver Description

Various mass driver designs were considered in an attempt to find the best one for the mission. Figure 10 below shows two such alternative designs that were investigated. Pictured on the left is a single one-way mass driver track powered by a large solar power array. On the right is a mass driver design which includes a deceleration section of track utilizing re-usable "buckets" that hold the payload during launch.



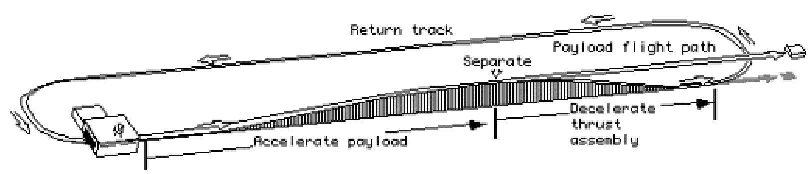


Figure 10: Sample Mass Driver Architectures

The mass driver 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 main components of the system are super conducting wire and silicon-controlled rectifiers. 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 11 below. All mass, power and cost estimates are based on the work of the late G. K. O'Neill of Princeton University [7]. The first rows of the table are design dependent variables. 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.

The remaining rows in Table 11 are outputs based on the design variables. Total system mass includes both the mass driver system as well as the power generating and storage facilities. The annual recurring cost accounts for costs associated with each launch as well as track maintenance. Non-recurring cost accounts for DDT&E, TFU, transport of system to lunar surface, track construction, power generation and power storage facilities.

Table 11: Baseline Mass Driver System Requirements

Mass Driver	
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

3.1.4 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 11.

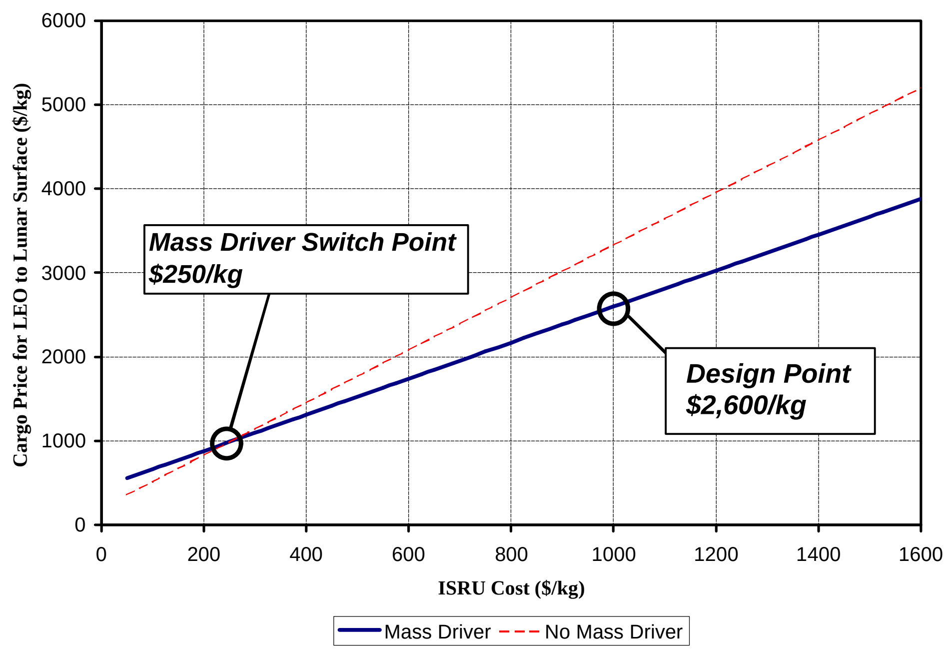


Figure 11: Customer Price as a Function of ISRU Cost

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 as described in Section 2.4.2, a cost breakdown was found for the baseline vehicle as shown in Table 12. 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 12: 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

3.2 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 kg 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 500s, the cost per kilogram for ISRU production is brought to $50, and a 20% technology reduction factor (TRF) used.

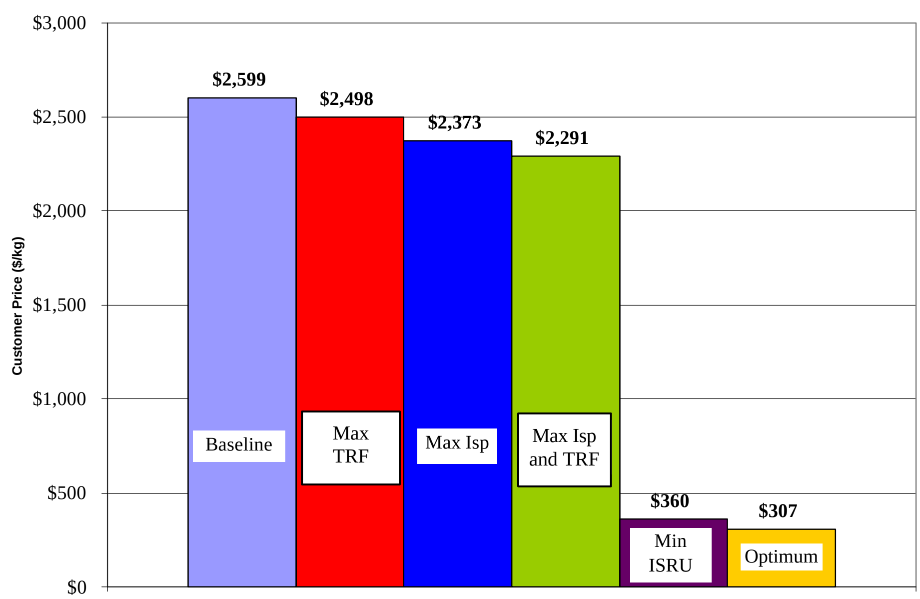


Figure 12: Price to charge customer per kg of payload for the optimal and baseline design cases

A comparison of different designs to the baseline vehicle is shown in Figure 12. 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 13 shows how sensitive the price to charge the customer is to the cost of ISRU propellant production.

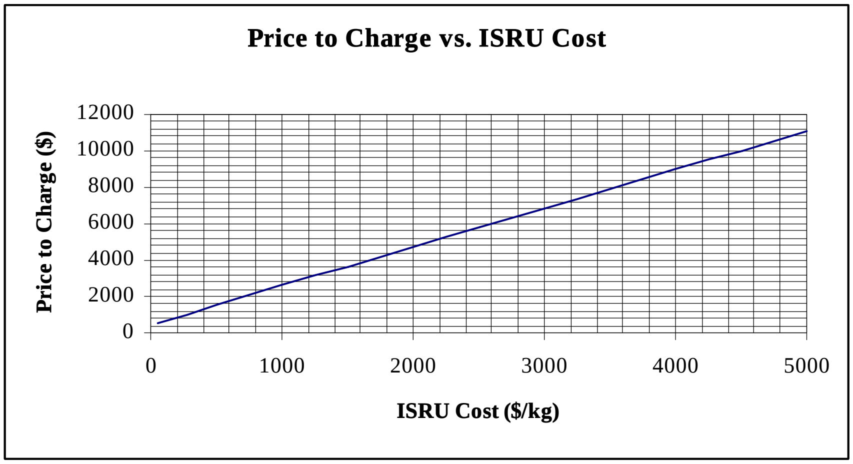


Figure 13: Effects of ISRU Cost on the Price to Charge the Customer

3.3 Independent Verification of Results

As discussed in Section 2, a Review Team paralleled the work of the Design Team throughout the design process. The Review Team used the same architecture but completed an independent analysis of the vehicle. The Review Team used more conservative values such as a different schedule requiring more vehicles, a heavier aerobrake based on current technology, and certain assumptions for the cost estimation such as a higher complexity factor for the RCS system. DDT&E for the main engine was also included in this verification analysis. The results of this analysis were within 13% of the price determined for the baseline case by the Design Team.

4. 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.

5. Outreach

The most important outreach operation of the MARTA Project is to the scientific community. By identifying a key driver in reducing the cost of an Earth-Moon transportation system, the MARTA team feels it has made an invaluable contribution to the exploration and commercial development of the Moon.

Additionally, the team has extended its outreach to the political arena. A MARTA team member lobbied members of Congress in March 2000 to help publicize the importance of utilizing space resources to provide access to space for everyone interested. MARTA has also been in contact with Lunar Prospector principal investigator Dr. Alan Binder about the MARTA design, learning from him some of the nuances of the results from Lunar Prospector and discussing his plans for commercial lunar exploration beginning with missions based on data purchase.

6. Acknowledgements

The members of the design team would like to express their utmost gratitude to Dr. Olds, for his guidance during every stage of the design. The project could not have taken place without the invaluable experience he helped pass on to us. We would also like to thank to those who have preceded us at Georgia Tech: Ashraf Charania, Dave McCormick, Irene Budianto, Laura Ledsinger, Dave Way, John Bradford, Kris Cowart, and Brad St. Germain for their support and advice throughout the project. Lastly, we would like to thank Dr. Alan Binder for providing us with previously unpublished data about the Lunar Prospector Mission and discussing his plans for commercial follow-on missions.

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