1-1-2015

Architecture Study For A Fuel Depot Supplied From Lunar Resources

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ARCHITECTURE STUDY FOR A FUEL DEPOT SUPPLIED FROM LUNAR RESOURCES

by

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A Thesis

Submitted to the Graduate Faculty

of the

University of North Dakota

in partial fulfillment of the requirements

for the degree of

Master of Science

Grand Forks, North Dakota
December
2015

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November 23, 2015
Date

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Title	Architecture Study for a Fuel Depot Supplied From Lunar Resources
Department	Space Studies
Degree	Master of Science

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Thomas M. Perrin
November 17, 2015

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ACKNOWLEDGMENTS

I express my sincere appreciation to the members of my advisory committee for their guidance and support during my time in the master's program at the University of North Dakota.

I also want to express my appreciation for my wife, Diana, and my daughter, Sarah, who kept the home fires burning while Dad spent many nights and weekends at the office working on "his space stuff".

I want to thank Mr. Charles L. (Les) Johnson, Deputy Manager of the Advanced Concepts Office at Marshall Space Flight Center. Mr. Johnson was keenly interested in the work and assisted me in locating subject matter experts when needed.

I also want to thank Mr. Patrick S. McRight, Manager of the Propulsion Systems Design & Integration Division at Marshall Space Flight Center. Mr. McRight tutored me on the complexities of cryogenic propellant storage and transfer, and frequently made himself available as a sounding board for my ideas.

Mr. Steven G. Sutherlin, a Propulsion Systems Engineer in the Marshall Space Flight Center Advanced Concepts Office, also contributed enormously to my effort. We discussed spacecraft design and its influence on propellant boiloff. Mr. Sutherlin also helped me wrestle with characterizing the thermal environment at different points in space, and checked my boiloff calculations with in-house tools he uses in his work. I am deeply grateful.

Mr. Larry Kos, a mission designer in the Marshall Space Flight Center Advanced Concepts Office, contributed his expertise regarding the thrust-to-weight ratio for lunar vehicles.

Mr. Robert Werka, a Senior Aerospace Flight Systems Engineer at Marshall Space Flight Center, contributed his expertise regarding rocket engine oxidizer-to-fuel ratios, as well as his friendship.

Lastly, I want to thank Dr. Alan Wilhite, Distinguished Langley Professor of Advanced Aerospace Systems Architecture at the Georgia Institute of Technology. While we only corresponded by email, Dr. Wilhite introduced me to the Modified Lockheed Model, and responded to my many questions as if I were one of his own students. Dr. Wilhite retired in December, 2014.

ABSTRACT

Heretofore, discussions of space fuel depots assumed the depots would be supplied from Earth. However, the confirmation of deposits of water ice at the lunar poles in 2009 suggests the possibility of supplying a space depot with liquid hydrogen/liquid oxygen produced from lunar ice.

This architecture study sought to determine the optimum architecture for a fuel depot supplied from lunar resources. Three factors – the location of propellant processing (on the Moon or on the depot), the location of the depot (on the Moon or in cislunar space), and if in cislunar space, where (LEO, GEO, or Earth-Moon L1), and the method of propellant transfer (bulk fuel or canister exchange) were combined to identify 18 potential architectures. Two design reference missions (DRMs) – a commercial satellite servicing mission and a Government cargo mission to Mars – were used to create demand for propellants, while a third DRM – a propellant delivery mission – was used to examine supply issues. The architectures were depicted graphically in a network diagram with individual segments representing the movement of propellant from the Moon to the depot, and from the depot to the customer.

Delta-v and time-of-flight information were developed for each network segment using restricted two-body techniques. Propellant expended was calculated using the rocket equation, while anticipated boiloff was calculated using the Modified Lockheed Model. Chilldown losses were also calculated with respect to bulk fuel transfer. The depot was assumed to have active cooling of cryogens, while the DRM vehicles were assumed to employ passive insulation only. Overall, propellant consumption and losses were calculated in moving propellant to the depot, or

in direct delivery to the customer vehicles. Similar consumption and losses were calculated for the customer DRMs in performing their missions and maneuvering to the depot or transfer location to refuel. The network diagram was then analyzed to determine which architecture satisfied the DRMs for the smallest mass of propellant.

The study concluded that shipping water in bulk to be processed into propellant on a depot at L1 consumed/lost the least mass of propellants. L1 is the most efficient fuel transfer location because of delta-v considerations, and shipping water to the depot avoids boiloff losses en route, and avoids chilldown losses between the tanker vehicles and the depot. For all candidate architectures, propellant boiloff in microgravity was less of a factor than anticipated, and was far overshadowed by delta-v requirements and resulting fuel consumption. Bulk fuel transfer is the most flexible for both the supplier and the customer. However, since canister exchange bypasses the transfer of bulk cryogens in microgravity and the necessary chilldown losses, canister exchange shows promise and merits further investigation.

CHAPTER I

INTRODUCTION

Background

Apollo-era mission design was based on taking everything needed for a mission from the Earth. This was an obvious choice. One reason behind this choice was the challenge by President Kennedy to land on and return from the Moon by the end of the decade. Another reason behind the choice had to do with the limited knowledge of the Moon and its resources. When traveling such a great distance from Earth into the unknown, it only made sense to take everything needed.

However, Earth's deep gravity well makes this paradigm expensive. It has been estimated the space shuttle cost $18,413/kg to place an object in low earth orbit (LEO) (London, 1994). Having to take all the fuel needed for a mission limits the size of the payload that can be taken. It would be far more cost effective to refuel vehicles in space, and the idea has been around from the very start of the space program.

The existence of lunar ice was first predicted in 1961 by Watson, Murray, and Brown in their paper *The Behavior of Volatiles on the Lunar Surface* in the Journal of Geophysical Research. They showed that water is actually one of the most stable of the lunar volatiles, and predicted that over the life of the Moon, water could have migrated to the cold traps at the lunar poles (Watson, 1961).

The idea lay dormant until 1979, when J.R. Arnold again suggested the presence of water on the Moon in his paper *Ice in the Lunar Polar Regions*, also in the Journal of Geophysical

Research. Arnold verified the stability of the lunar cold traps and the trapping mechanism and further advocated a lunar mission to search for ice deposits (Arnold, 1979).

In 1998, NASA launched the Lunar Prospector probe into lunar orbit. Included on board the probe was an instrument called a neutron spectrometer. The experiment searched for and confirmed the presence of hydrogen at the lunar poles which indicated the presence of ice (Spudis, 2011).

In 2009, NASA launched the Lunar Reconnaissance Orbiter (LRO) and Lunar Crater Observing and Sensing Satellite (LCROSS) missions. An Atlas V Centaur upper stage rocket was deliberately impacted into the Cabeus crater on October 9th, and the LCROSS spacecraft flew through the debris kicked up by the rocket. From the data gathered, NASA was able to confirm the presence of water ice. The size of the ice deposits has since been estimated to be as large as 600 million cubic meters (Spudis & Lavoie, 2011).

The confirmation of substantial deposits of water at the lunar poles suggests the Moon could provide liquid oxygen and hydrogen to an orbiting fuel depot. Such a plan represents an In-Situ Resource Utilization (ISRU) - based exploration paradigm – launch from the Earth using terrestrial resources, then use in-situ resources to refuel for the trip home…or to a more distant destination. A fuel depot would also enable/require the use of space vehicles tailored to specific applications – Earth-to-orbit vehicles, Moon-to-orbit vehicles, and in-space vehicles.

Consideration of an architecture for a fuel depot supplied from lunar resources gives rise to a great many questions. Where, for example, should such a depot be located? How many depots should there be? Should water harvested on the Moon be processed into liquid oxygen and liquid hydrogen on the Moon, or should it be shipped to the depot and processed on the depot itself? And how will the transfer of fuel be accomplished? Would it be better to ship in bulk and wrestle with the transfer of cryogens in microgravity, or would it be better to ship using standardized canisters, and refuel a customer spaceship simply by exchanging empty canisters for full ones? Regardless of the choices made, there will be a cost to both the supplier and the customer. Both parties will consume propellant in carrying out their respective missions, and both parties will lose propellants due to boiloff. So the task becomes choosing the architecture which promises to be the most efficient in terms of propellant consumption and loss.

Research Question

Which architecture satisfies the Design Reference Missions (DRMs) for the least amount of liquid oxygen (LO2) and liquid hydrogen (LH2) consumed in flight or lost due to boiloff?

Thesis Statement

Positioning a fuel depot in geostationary orbit would most efficiently enable the servicing of customers using propellant from lunar resources for the proposed Design Reference Missions.

Literature Review

At the outset of the thesis effort, a literature review was undertaken in two major topic areas. The first of these was fuel depots and propellant storage. Here the desire was to understand current thinking with regard to propellant depots, and to also understand the state of technology with regard to cryogenic propellant storage. The second major topic area was fuel depot architecture studies. The desire here was to understand current thinking about depot architectures – that is to say, how a depot or depots would be employed and support operations in cis-lunar space and elsewhere. Of specific interest was to see the number of depots called for, and the recommended locations. It was anticipated that writers would recommend a depot in GEO, since this is the location of many potential “customer” satellites.

Fuel Depots and Propellant Storage

The idea of constructing a fuel depot in space has been discussed almost from the beginnings of the United States’ space program. Stemming perhaps from American experience with the automobile, the idea of refueling in space was a reasonable assumption, and depictions of space depots and space stations have been in space art almost from the beginning.

But a more serious look at fuel depots requires more serious questions. What individual tasks must it be capable of performing? How would it be constructed? What technologies are needed?

It is useful at the outset to consider what kinds of tasks a depot would have to perform. Many authors appear to write for a narrow audience, assuming their readers already have some background in the topic. For example, Dallas Bienhof lists “Mature cryo fluid management capability” as a step toward establishing a propellant depot, but never explains what fluid management involves (Bienhof, 2007, p.10). Johnson does somewhat better. He points out that cryogenic propellant storage and transfer (CPST) is ranked number two of the top ten propulsion challenges facing NASA (Johnson, Meyer, Palaszewski, Coote, & Goebel, 2013). Johnson speaks of technology challenges, but the terminology he uses alludes to two key tasks the depot must perform – storing cryogenic propellants (liquid hydrogen and liquid oxygen) without significant loss to boiloff, and transferring propellant from the depot to a customer vehicle in microgravity. Howell lists four key tasks as necessary for a depot: supply vapor-free cryogenic liquids to an orbital transfer vehicle, perform mass gauging (i.e., measure how much propellant is in a given tank), store propellants with minimal or zero boiloff, and perform a leak-free fuel transfer (Howell, Mankins, & Fikes, 2006).

William Notardonato of NASA's Kennedy Space Center provides perhaps the most comprehensive list. In addition to the tasks already mentioned, he suggests the depot should perform electrolysis of water and the liquefaction of the resulting hydrogen and oxygen. He also notes the depot must perform power generation, perform active thermal control (i.e. actively cooling the cryogenic propellants), perform storage and distribution of propellants, and provide its own propulsion and maneuvering capability (Notardonato, 2013).

Lastly, Howell also adds several enabling technologies or tasks, such as performing teleoperated or fully autonomous operations, and performing in-space assembly, maintenance, and servicing (Howell et al., 2006).

How a depot could be constructed is another topic of interest. The literature reflects two basic schools of thought. One school recommends the launching of dedicated hardware that would be assembled on orbit, while the other school recommends repurposing spent rockets or other existing components of flight hardware to assemble a depot. Bienhof, for example, details a Boeing concept for a low Earth orbit propellant depot. The concept uses a hub and spoke configuration, with a central truss structure as the hub and individual propellant "tank sets" radiating outwards. Both the truss structure and the tank sets would be specifically designed and constructed for the depot, launched into orbit, and assembled. The tank sets and truss structure would not be repurposed from existing flight hardware (Bienhof, 2007).

Honour, Kwas, O'Neil, & Kutter (2006) suggest a concept whereby the forward end of an Atlas V Centaur would be mated to the aft end of a modified Atlas V Centaur. They write,

> The modified Centaur would consist of an elongated LH2 tank connected to a small boiloff storage tank. Both the Centaur and modified Centaur would be encapsulated within the Atlas V 5-meter payload fairing at launch. Once on orbit, residual LH2 within Centaur would be transferred to the modified Centaur; the residual H2 would be purged with Helium. The Centaur would then be refilled, on orbit, with L02. Consequently, the modified Centaur functions as the on-orbit LH2 storage module, and the Centaur functions as the on-orbit L02 storage module. The dual propellant … concept has the advantage of being able to store both LH2 and L02. Further, the concept utilizes existing, or slightly modified, flight hardware. (Honour, Kwas, O'Neil, & Kutter, 2012, p.2)

United Launch Alliance (ULA) carries the repurposing idea a step further. Zeglar, Cutter, & Barr (2009) report that United Launch Alliance (ULA) is developing a common propulsion stage called ACES – Advanced Common Evolved Stage – based on its experience with the Centaur and Delta rockets. ACES is designed with the express intent to be reused after achieving orbit. It has no helium- or hydrazine-based systems. It can be produced in different lengths. All pressurization, attitude control, and power generation are based on the consumption of its main propellants, and ACES is designed to be refueled in space. Two ACES stages are mated end-to-end to form the depot, and a passive sun-shield is deployed around the liquid hydrogen tank (Zeglar, Cutter, & Barr, 2009).

Looking at planned technology development and/or technology demonstrations also provides some insight into the technologies needed for fuel depots. Meyer, Motil, Kortes, Taylor,

& McRight (2012) provide an excellent list of twelve CPST-related technologies. Most of these are ranked at Technology Readiness Levels (TRL levels) of 4-6 (i.e. prototyping in a laboratory or relevant environment). They need further development and/or system-level demonstration before they could be considered mature enough for operational use (Meyer, Motil, Kortes, Taylor, & McRight, 2012). Their list – with short explanations or comments – includes:

1- Active thermal control: Cryocoolers technology (Cryocooler is the name given to refrigeration systems used to keep cryogenic propellants cold. Since liquid oxygen and liquid hydrogen are very cold, 80 Kelvin and 20 Kelvin, respectively, even the best cryocoolers are not very efficient.)
2- Thick multilayer insulation (MLI) with foam substrate. (Multilayer insulation is used to reduce the amount of heat entering fuel tanks. Spray-on foam insulation (SOFI), such as that used on the Space Shuttle external tank, has no insulating value in space.)
3- Low conductivity structures: high strength composite struts (If metal struts were used to support a cryogenic fuel tank, the struts would be pathways for heat to enter the tank.)
4- Microgravity pressure control: thermodynamic vent system (If cryogenic propellant begins to boil inside a tank, the gaseous propellant must be vented before the increased pressure inside the tank causes the tank to rupture.)
5- Microgravity pressure control: Mixing pumps (Mixing pumps inside a cryogenic propellant tank are used to assist in keeping the temperature of the propellant as uniform as possible.)
6- Unsettled liquid acquisition devices (A liquid acquisition device (LAD) is a metal structure – often with vanes and metal screens -- inside a cryogenic propellant tank

that takes advantage of the surface tension of the liquid propellant to "wick" the propellant and direct it toward the throat of the tank. An unsettled LAD would be one designed to function in an unsettled tank – a tank in which no force had been applied to "settle" the propellant.)

7- Microgravity transfer line chilldown (Chilldown is the term given to cooling a transfer line by intentionally filling it with liquid propellant and allowing the propellant to boil off, thus cooling the line. Chilldown of the transfer line is one of the first steps taken to prepare for the transfer of cryogenic propellant from one tank to another.)

8- Pressurization systems. (Pressurization systems reduce or prevent the boiloff of cryogenic propellants, and could be used to settle the propellant.)

9- Settled mass gauging (Mass gauging is measuring the mass of propellant in a settled fuel tank in microgravity.)

10- Unsettled mass gauging (Unsettled mass gauging measuring how much propellant is in an unsettled tank in microgravity.)

11- Microgravity chilldown tank (Methods to chill down a fuel tank in microgravity prior to filling it with cryogenic propellant. If propellant begins to boil off as soon as it enters a tank, the resulting pressure will inhibit the tanking process.)

12- Automated leak detection (Detecting and locating leaks in microgravity before valuable propellant is lost.)

Fikes, Howell, & Henley (2006) also discuss technology developments for cryogenic fluid settling and acquisition. Most techniques for settling propellants in microgravity involve imparting some kind of acceleration to the propellants to drive them to the desired part of the tank. Fikes mentions several techniques – gravity gradient forces, surface tension, and rotation (rotation of propellant tanks or even the depot itself could impart centrifugal acceleration to the propellants). Perhaps most interesting, they also list tank exchange (Fikes, Howell, & Henley, 2006, p.7). Rather than trying to transfer fluid from one tank to another in microgravity, tank exchange would involve the customer spacecraft swapping empty fuel tanks for full ones. This, of course, would mandate some level of standardization of tank sizes and connecting hardware among all vehicles concerned.

Plachta and Kittel (2002), both NASA employees, examine cryogenic storage for conceptual orbit transfer vehicles. They predict the performance of a zero boil-off (ZBO) cryogenic storage system and then compare it to traditional, passive-only concepts. (The use of cryocoolers to actively chill on-board cryogenic propellants brings an added mass to the spacecraft for the cooling equipment, and an added power requirement for the spacecraft, which may require added mass (solar arrays, and so forth) to satisfy. A concern in employing an active system is the issue of mass savings. That is, the increase in the mass of the spacecraft must be offset by the reduction in propellant lost due to boil-off.) The results of Plachta and Kittel's modeling showed an overall mass savings in less than a week for liquid oxygen, two weeks for liquid methane, and approximately two months for liquid hydrogen. This means that when ZBO techniques are employed, a given mission would not have to carry additional propellant – over and above mission requirements – to compensate for the expected boiloff (Plachta & Kittel, 2002).

Zeglar et al. (2009) argue that active cooling and other cryogenic fluid management (CFM) technologies are not necessary. They argue that vaporized hydrogen can be used in two ways: 1) to suppress LO2 boiloff by removing heat, and 2) to be used as part of a solar-thermal propulsion system for reboost, station-keeping, and maneuver control (Zeglar, et al, 2009). These “losses” are viewed simply as a cost of doing business in LEO, and they view the simplicity of their proposed depots as worth the loss of hydrogen. They state, “Striving to suppress heating to the lowest possible level with exotic technology is pointless. Amplifying throughput is the best way to make the depot more efficient.” (Zeglar et al., 2009, p. 17)

Honour reminds us that we should not discount passive techniques, such as the use of sun-shades to limit the direct heating of propellant tanks by the sun. Passive techniques are often less expensive than more exotic solutions (Honour et al., 2012).

To summarize, fuel depots in space offer great promise to bring great change to space travcl. If and when established, fuel depots will need to accomplish two types of tasks. First, the depot must perform those general tasks associated with many spacecraft – power generation, attitude maintenance, orbit maintenance/station keeping, telemetry, and so forth. Second, depots will have to perform a number of depot-peculiar tasks – docking with a supplier vehicle or customer vehicle, accepting the transfer of cryogenic propellants from a supplier or transferring those propellants to a customer. Mass gauging – measuring the mass of propellant inside a fuel tank – is also a key depot task.

There are essentially two schools of thought regarding the construction of fuel depots – designing and constructing the depot as unique flight hardware and launching it from the Earth as the payload(s) of other rockets, or repurposing existing flight hardware and/or spent launch vehicles.

Although the idea of a space depot spurs the imagination, cryogenic fluid management (CFM) technologies – the storage and transfer of cryogens in microgravity -- are not considered mature enough for operational use. While liquid oxygen and liquid hydrogen are prized for their relatively high specific impulse, the loss of these cryogens in space due to boiloff is a major concern. Passive insulation techniques are the norm and cannot be ruled out, but many authors believe that active cooling is the only way to achieve acceptable levels of boiloff, and point to an overall mass savings for the spacecraft when active cooling is used. Technology development and demonstration efforts are underway in NASA and within industry to mature active cooling and the full range of CFM technologies.

Fuel Depot Architecture Studies

The second major topic area was fuel depot architecture studies. Beyond just looking at the depot itself, there is a need to understand the depot(s) as part of a larger system. For example, what is the area of operations served by the depot? How many depots are needed? Where are they located? Where is electrolysis performed? What are the other elements of this larger system? And perhaps most importantly, what measure or measures of goodness will be used to judge whether or not an architecture is effective, or to judge a candidate architecture against other candidate architectures?

Even if unstated, the writers surveyed agreed the area of operations is cis-lunar space. Cis-lunar space is understood to be that area of space between the Moon and the Earth – although some writers also include the Earth-Moon Lagrange Point L2 on the far side of the Moon. There are likely several reasons for this area of operations. First, with the exception of planetary probes, man has not attempted significant operations beyond our Moon. Second, at the time in which many of the most recent papers were written, NASA’s Constellation Program was underway with its focus on a return to the Moon. Lastly, the area in which the most commercial space activity takes place is right around the Earth. Horsham, Schmidt, and Gilliland point out that the LEO-to-GEO region "is the only accessible, extraterrestrial region with both near and far-term civil, military, and commercial development potential." (Horsham, Schmidt, & Gilliland, 2010) At first this might seem a profound statement, but it is not. Rather, it is just their simple recognition of the LEO-to-GEO region as where the majority of customer satellites are located.

The next three questions – how many depots are needed, where are they located, and where is electrolysis performed – overlap. Duke, Diaz, Blair, Oderman, & Vaucher (2003) describe two potential depot architectures supplied from the Moon. The first architecture employs two fuel depots, one located at L1 and the other in low Earth orbit. Tankers deliver lunar water to the L1 depot, which produces enough LO2/LH2 to send the tanker back to the Moon, and a second tanker with water to the depot at LEO. Water processed into fuel at LEO is used to fuel an orbital transfer vehicle (OTV) which lifts a customer's satellite from LEO to GEO. The remaining fuel is used to fly the tanker back to L1.

Their second architecture is simpler, and includes a single fuel depot located at L1. In this architecture, electrolysis is performed at L1 as before, but the OTV is based at L1 rather than LEO. The OTV receives fuel, flies to LEO and boosts the customer's satellite to GEO, then flies back to L1 (Duke et al., 2003).

In another paper, Zeglar et al. (2009) of United Launch Alliance (ULA) address an architecture that functions to support lunar operations. Written in the shadow of the Constellation program, they describe two depots, one located in LEO, and the other located at Earth-Moon L2. Fuel for the LEO depot would be supplied from the Earth, and a portion of that fuel would be "pushed" forward to the depot in L2. They recognize that LEO is thermally stressing for a depot, while the L2 location is considered to be near ideal. Fuel from the LEO depot is used to support cis-lunar operations, while fuel from the L2 depot is used to fuel lunar landers (Zeglar et al., 2009).

Richard Oeftering of NASA's Glenn Research Center describes a more bootstrap approach to a fuel depot supplied from lunar resources (Oeftering, 2012). Oeftering's plan starts with an electrolysis/liquefaction facility on the Moon which also functions as the fuel depot. The depot delivers propellants directly to customers using small tanker vehicles. As demand grows, the facility grows with it, and he proposes to graduate to larger and larger tanker vehicles, until such time as it is determined that a true orbiting depot is needed.

Oeftering rejects the idea of a LEO-based depot. He states, "Locating a depot in LEO seems obvious since that is where the users are. However, the LEO thermal environment is not favorable to cryogenic storage. Further, phasing and orbital plane inclination changes near Earth are particularly inefficient." (Oeftering, 2012, p.6) Instead, Oeftering chooses to place his depot at L1 for both the thermal and delta-v (Δv) advantages.

Oeftering discusses two uses for the depot – to facilitate the servicing of satellites in geostationary orbit, and to refuel vehicles departing for Mars or Near Earth Objects (NEOs), but he does not create design reference missions that would quantify the demand for propellants. He recognizes that demand will likely start small and grow over time, but the lack of design reference missions prevents him from addressing this with any specificity. The same is also true with respect to cryogenic boiloff. He recognizes that L1 is a better (colder) location for a depot than LEO, but makes no attempt to quantify expected losses.

Notardonato proposes a fuel depot in low earth orbit, but with a twist. He calls his depot concept a "propellant production and liquefaction spacecraft (PPLS)." (Notardonato, 2012, p.238) As mentioned earlier, the PPLS would be supplied with pure water. It would electrolyze the water and would liquefy and store the captured oxygen and hydrogen. Notardonato reasons that since water is much denser than liquid hydrogen, payload mass fraction will be larger, resulting in reduced launch costs. He also notes that ground support equipment would be less complex, further reducing costs. However, there are trades. In his concept, he calculates that solar arrays similar to the International Space Station would be required, generating 65 kilowatts of power. Furthermore, the electrolysis of the water into hydrogen and oxygen, followed by the liquefaction into liquid oxygen and liquid hydrogen, would not be rapid. He estimates that his vehicle would take 6 months to process 20 metric tons (20,000 kg) into usable propellant (Notardonato, 2012).

Horsham, Schmidt, and Gilliland (2010) describe a more grand approach to depots in their 2010 paper. They propose a so-called "space harbor" to be assembled in LEO but then boosted to an unspecified higher orbit. The harbor would serve as an operational platform for as many as 16 different servicer spacecraft providing a number of in-space services. The harbor would also include a fuel station, or depot. The servicer spacecraft could be owned and operated by private companies or even governments (Horsham, Schmidt, & Gilliland, 2010).

The space harbor would be a "place to call home" where the spacecraft are refueled and repaired, or otherwise serviced to prepare them for subsequent missions. Like the ISS, the space harbor itself would be supplied by numerous Earth-to-orbit launches.

They also recognize that any sort of LEO-to-GEO satellite servicing capability will necessarily be "the domain of semi-autonomous (i.e., teleoperated) and fully-autonomous (i.e., artificially intelligent) robots." (Horsham et al., 2010, p.2). This stems from the distances involved and the complexities of human spaceflight.

The various papers examined share a few common elements beyond the depot itself. First, there must be some sort of propellant processing facility from which the propellants are shipped. Many writers (Horsham, Zegler) see this facility as being on the Earth, and point to the relative ease in establishing and operating the facility, as compared to a facility on the Moon or in orbit. Other writers (Oeftering) see potential advantages to processing propellant on the lunar surface. Lastly, some see processing the propellants on board the orbiting depot itself (Oeftering, 2013; Notardonato, 2012; Duke, 2003).

Many writers mention the use of tanker vehicles. Although unstated, the implication is the tanker vehicles are larger and deliver fuel from the processing facility to the depot(s), while smaller orbital transfer vehicles (OTV) deliver propellants from the depot to the customer. Duke states the tanker must be capable of launching from the Moon, delivering its payload to the depot, and returning to the Moon (Duke, 2003). This contrasts with the OTV, which never lands on the Moon. In Duke's second architecture, in which there is only one depot, the functions of the tanker and OTV are combined in a single vehicle.

Measures of goodness with which to judge the relative merit of candidate space depot architectures are elusive, primarily for the lack of data. One example of this elusiveness is cost. Duke et al. (2003) points out there are presently no customers for propellant delivered in space What he means is that any entrepreneur seeking to establish a space fuel depot takes an enormous financial risk that customers might fail to materialize. Indeed, no rational company would embark on such a venture without some assurances the business case was solid. The use of a fuel depot represents a new paradigm, a new way of doing business….and success is not guaranteed. Another problem with judging cost is the lack of existing data. Since a fuel depot has not been attempted before, it is difficult to measure the actual construction or operations costs.

Looking at cost from a different perspective, Bienhof refers to a 2005 speech by [then] NASA Administrator Dr. Michael Griffin, in which Griffin estimates the value of propellant in low Earth orbit (LEO) to be $10,000 per kilogram. Bienhof uses that value as a benchmark, and argues the operation of a fuel depot in LEO must be able to deliver propellant at or below that price in order to be financially viable (Bienhof, 2007). Bienhof implies, of course, that he would expect NASA to be the primary customer for the depot.

Duke et al. (2003) also points to the difficulty in assessing the costs of a depot. He attempts to construct a financial model for each of his two architectures, and concludes there are numerous variables for which there are simply insufficient data. These variables include the quantity and quality of ice in the lunar regolith, information about the machinery that would be needed to excavate the ice, how electrical power would be generated, and how the fuel produced would be transported to the depot, among others.

Zeglar et al. (2009) points out the benefit of a fuel depot in shaping the infrastructure on Earth for launch vehicles and other launch resources. He argues that if regular transportation is needed beyond LEO, a fuel depot would facilitate the use of smaller, less expensive launch vehicles. Using lots of smaller launch vehicles would lead to "high infrastructure utilization, economic production rates [economies of scale], high demonstrated reliability, and the lowest possible costs." (Zeglar et al., 2009, p.1)

Another measure of goodness might be throughput – how much propellant a given architecture could produce and deliver in a period of time. Throughput speaks to the relative efficiency of one candidate architecture versus another. This is not addressed directly by any of the authors, although Zeglar and his colleagues mention throughput in their argument against using expensive cryocoolers to minimize boiloff.

Simplicity could be another measure of architecture goodness, but is not addressed directly by the authors. For example, simplicity might be defined and measured as the architecture that uses the fewest number of vehicles. Duke et al. (2003), for example, describes two architectures – one with two depots and another with only a single depot. Even a casual reading of their descriptions reveals the latter architecture as having the fewer vehicles and being more straightforward. Simplicity might also be defined by the number of steps or transfers a given architecture requires to produce and deliver propellants. Still another facet of simplicity might be that all the vehicles in a given architecture use the same LH2/LO2 propellant. They imply this in describing the operation of his lunar water tanker, noting that, "The vehicle is capable of landing near the propellant production plant, taking on a payload of water and cryogenic propellants and traveling from the Moon to the L1 propellant depot." (Duke et al., 2003, p.1221)

Fuel consumed or lost to boiloff could also be a measure of the relative goodness among competing architectures. Although none of the authors address this directly, several acknowledge the poor thermal environment in LEO, i.e., would experience the greatest rates of boiloff, and recommend stationing a depot at L1 or L2. Oeftering and Duke make mention of Δv requirements, and recommend the use of aerobrakes or similar devices to reduce the Δv required to deliver fuel to low Earth orbit, so they are at least mindful that delivering propellants is not without these operational costs. Oeftering, in particular, goes to some length to describe the use of the aeroshells in his architecture, which would be detached after propellants have been delivered in LEO and flown back to the Moon separately (Oeftering, 2011). But the idea of looking at fuel consumed or lost has some merit, primarily because there are more knowns than with other possible measures.

In summary, the fuel depot is not a stand-alone entity, but must function as a part of a larger system (architecture) involving other space vehicles. The area of operations for such an architecture, at this point in man's spacefaring history, is likely to be cis-lunar space, primarily because the majority of existing space assets are there and that is where the demand for in-space services such as refueling is likely to be.

The number of elements in the architecture may vary with the maturity of the architect-true. If fuel is processed on the Moon, the Moon facility may also serve as the depot early on, and tanker vehicles might deliver propellants directly to customers. A more mature architecture might have a fuel processing facility, but with tankers delivering fuel to an orbiting depot, and perhaps smaller orbital transfer vehicles delivering fuel from the depot to the customer.

Most authors speak of a single depot, while some speak of two. While customer satellites and other vehicles will mostly be at GEO or below, locating a depot at Earth-Moon L1 is driven by Δv considerations and a more favorable (colder) thermal environment. LEO is acknowledged to be a poor thermal environment for a fuel depot.

Judging the relative merit of candidate architectures is a difficult task, primarily due to the lack of relevant data. Potential methods to judge candidate architectures include monetary cost, throughput, simplicity, and propellant consumption and loss.

CHAPTER II

METHODOLOGY

General Intent

The general intent of this thesis is to create candidate architectures for a fuel depot supported from lunar assets, and to evaluate those architectures on the basis of the mass of propellant consumed and the mass of propellant lost due to boiloff. There are numerous steps necessary to accomplish this. These steps are illustrated in Figure 1 and are described below. The discussion here provides an overview of the methodology. Detailed discussions of specific tasks (calculating fuel consumption or boiloff, etc.) are provided in subsequent chapters.

Step-by-Step

The first step in the methodology is to establish ground rules and assumptions. These are necessary to bound the problem being attempted and to make it more manageable. A complete list of ground rules and assumptions is provided in Appendix B of this document.

The next step is to define candidate architectures. It is anticipated from the start that candidate architectures will be defined by several attributes, including the proposed location of the depot, the location where electrolysis will be performed, and the method of fuel transfer.

We then depict the candidate architectures as a network diagram. Each candidate architecture is unique, and the choices made in developing the architecture can be depicted as a separate path in the diagram. The diagram is an excellent method of depicting the candidate architectures, and visualizing relationships.

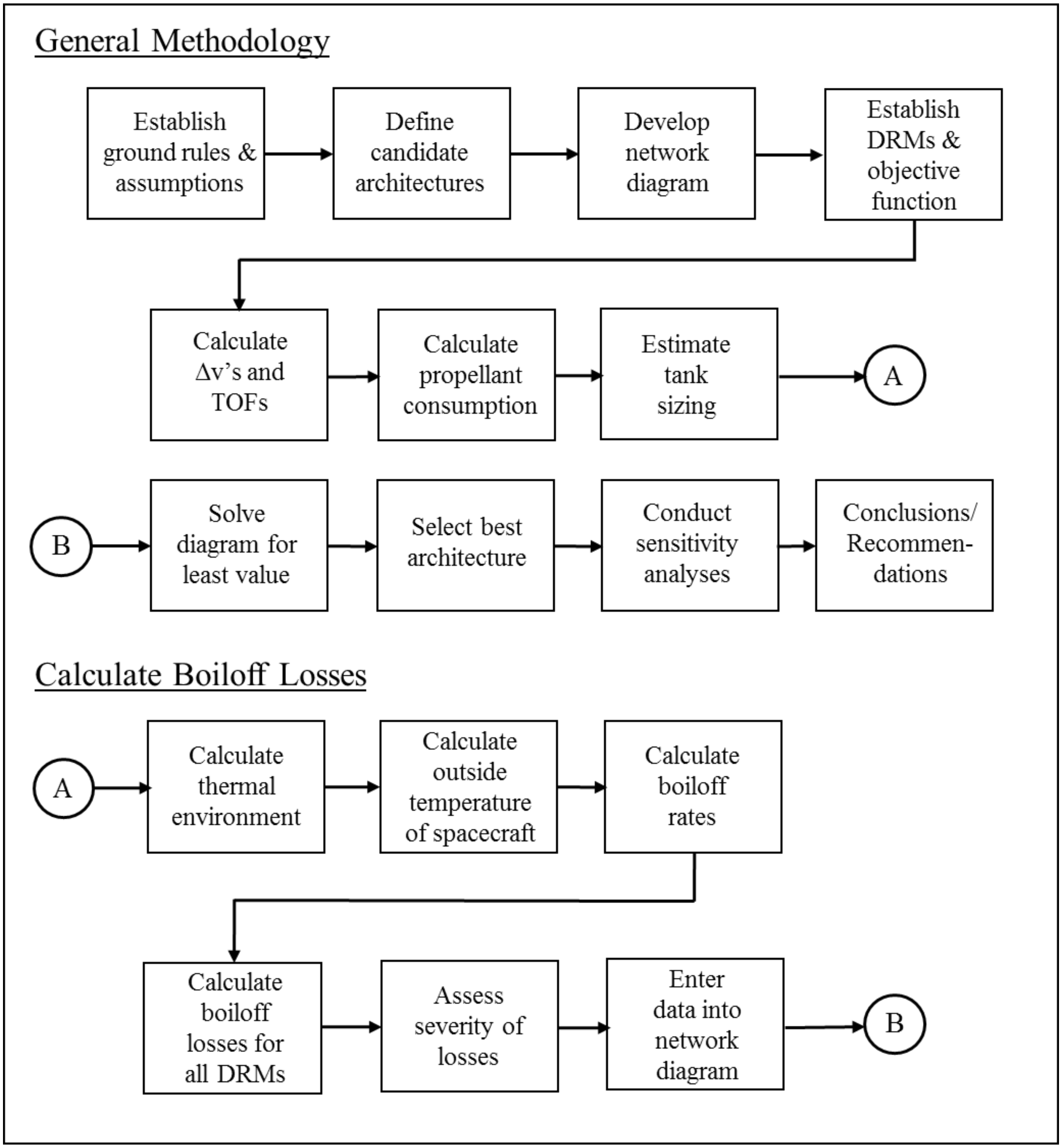

Figure 1. Methodology

The next step in the methodology is to establish the design reference missions (DRMs) and the Objective Function. The design reference missions will include detailed descriptions of the customer(s) for the depot, as well as the supplier that brings the fuel to the depot. The

Objective Function is a mathematical expression for what the thesis is attempting to accomplish. The research question asked, "Which architecture satisfies the Design Reference Missions (DRMs) for the least amount of liquid oxygen (LO2) and liquid hydrogen (LH2) consumed in flight or lost due to boiloff?" This implies the objective function will be a minimization function, and the individual terms in the function will be measurements of propellant consumption or loss.

After establishing the DRMs, we want to calculate the change in velocity (i.e., Δv) and the time-of-flight for a number of maneuvers in cis-lunar space. For example, if a vehicle departs the Moon and delivers fuel to a customer in geostationary orbit (GEO), it is important to know the Δv needed to perform the maneuver and also the time of flight. These values will be calculated using restricted two-body techniques.

The Δv values will then be used to calculate propellant consumption. The rocket equation will be used to calculate how much fuel will be needed to accomplish the design reference mission tasks. Then, the rocket equation will be used again to calculate how much propellant the supplier will need to deliver propellant to the customer(s). This information will have many uses. For example, it can be used to calculate the propellant tank sizes for all of the vehicles concerned, which is the next step in the methodology. But, together with information from the DRMs, it gives insight into how much propellant will be needed over time, and the capacity of the depot.

Calculating propellant tank sizes takes several steps. Initially, the use of the rocket equation permits the calculation of a final mass of a vehicle, based on the initial dry mass, the specific impulse of its engine, and the Δv required. Subtracting the initial mass from the final mass gives the amount of propellant needed. But this is not the whole story. We then need to break down the propellant mass into the mass for liquid oxygen and the mass for liquid hydrogen. This is accomplished by assuming a 6:1 oxidizer-to-fuel ratio, which is common for that kind of rocket engine (Huzel and Huang, 1992). (The science behind the 6:1 ratio is described in detail in Chapter VIII.) For example, 14,000 kilograms (kg) of propellant would break down to 12,000 kg of LO2 and 2,000 kg of LH2. Dividing the resulting masses of LO2 and LH2 by their respective densities gives the desired volumes of the fuel tanks.

At this point, it is necessary to calculate expected losses of propellants due to boiloff. Several tasks are involved. The first task is to characterize the thermal environment in which a given spacecraft must operate. This is done by calculating the heat load on the spacecraft in different orbits. The heat load consists of solar flux, reflected earth heating, and Earth-infrared heating. Second, these values are used to calculate the outside temperature of the spacecraft. Third, the outside temperature of the spacecraft and the size and configuration of the propellant tanks are used to calculate a boiloff rate. Fourth, the anticipated boiloff is calculated based on the length of time the spacecraft is exposed to that thermal environment – taken from the times of flight calculated for the individual maneuvers as described earlier.

After calculating the boiloff losses for each of the DRMs, we have to look to see whether the vehicle has enough fuel remaining to accomplish its mission. We do this by subtracting the boiloff losses from the fuel volume, and comparing the amount remaining to the amount of fuel estimated for the DRM vehicle to perform its task. If the amount of fuel lost to boiloff is too great, then the size of the fuel tank must be increased to compensate for the anticipated losses.

It is important to point out that these steps, i.e., calculating fuel consumption, calculating boiloff, etc., must be performed for every DRM vehicle for every candidate architecture. Also, supply vehicles that deliver fuel can experience boiloff from the propellant they use for their own propulsion, as well as from the propellant which is their payload. Then, this information is assembled so the candidate architectures can be compared.

The architecture with the smallest overall values for combined propellant consumption and loss ostensibly will be the “best” architecture. However, it is expected that there will be a number of lessons learned from the exercise. Some statistics will be computed – propellant losses as a percentage of propellant consumed for each vehicle, propellant losses as a percentage of propellant consumed across all vehicles, and propellant losses as a percentage of fuel shipped.

CHAPTER III

CANDIDATE ARCHITECTURES

Ground Rules and Assumptions

At the outset of the thesis research, a number of ground rules and assumptions were identified and captured. Several of these were dictated by tools at the author's disposal, such as the use of restricted two-body techniques to calculate the Δv and time-of-flight associated with different spacecraft maneuvers, and assumption of impulsive vehicle accelerations. Other assumptions such as assuming circular, coplanar orbits were used to simplify calculations without "assuming away the problem"; that is, to make the effort more manageable and yet still obtain insights into the fuel depot topic.

The complete listing of ground rules and assumptions is found in Appendix B: Register of Ground Rules and Assumptions. A number of constants are used in the calculations. These are provided in Appendix C: Dictionary of Constants Used. Lastly, formulas used in this thesis are recorded in Appendix D: Glossary of Formulas and Variables.

Parameters Chosen

The topic of space fuel depots has been discussed in scientific literature for years. It is widely accepted that being able to refuel after launching into Earth-orbit reduces the amount of fuel that must be launched with the rocket, and enables larger payloads to be taken to more distant locations such as the Moon. So the discussion is often not why a depot or depots might be useful, but rather how to go about it.

But the vast majority of literature assumes the depot or depots would be supplied from the Earth. The prospect of supplying a depot from resources mined at the lunar poles brings to mind a number of questions. Certainly, such an operation would require a massive investment in lunar infrastructure and propellant processing capability. But that is not the focus of this research effort. Instead, we focus on four factors:

1- Since the fuel being supplied from the Moon will be liquid oxygen (LO2) and liquid hydrogen (LH2) from lunar ice, where should electrolysis and liquefaction be performed – on the Moon or on the orbiting depot itself? One argument for processing the water on the depot is the ease with which water (not a cryogen) could be shipped to the depot without fear of boiloff.

2- Where should the depot be? Locations frequently mentioned in literature include LEO, geostationary orbit (GEO), and at the Earth-Moon Lagrange Point L1, located between the Moon and the Earth. Locating the depot on the Moon is also an alternative.

3- Where should the transfer to the customer take place? The simple answer to this would be "at the depot", except in the case in which the depot is on the Moon. If the depot is on the Moon, the direct transfer from tanker to customer vehicle would be required.

4- How should the fuel be transferred to the depot, or from the depot to a "customer" vehicle? While the transfer of cryogenic fluids in microgravity has been studied, it has never been performed in space in any significant quantity (Chato, 2005). This gives rise to the notion of bypassing the fluid transfer altogether by exchanging the fuel tanks themselves – swap an empty tank for a full tank – much the same as is done with gas grills on the Earth. The factors chosen for examination in this thesis are shown in Table 1 below.

Before progressing further, it is pragmatic to ask whether the canister exchange method is valid. That is, the propellant tanks for a given rocket are sized to provide the proper ratio of propellant and oxidizer for the desired level of performance. For liquid hydrogen/liquid oxygen rocket engines, this ratio is called the oxidizer-to-fuel ratio, or O/F ratio. For these engines, the desired O/F ratio is usually 5.5 - 6.0:1 (Huzel and Huang, 1992). Can this still be accomplished if the propellant tanks are “standardized” at a given volume? The answer is “yes” – standardized propellant tanks can meet the desired O/F ratio easily.

Table 1. Architecture Defining Parameters and Potential Values

Parameter	Possible Values	Remarks
Location of depots	On Moon, L1, GEO, LEO	Locations most frequently mentioned in technical literature.
Location of electrolysis/ liquefaction	On Moon, On-board orbiting depot	Electrolysis is performed daily in microgravity onboard the ISS. The technology is suitable for scaling.
Location of fuel transfer to customer	L1, GEO, LEO	Transfer at depot location, except for Moon.
Method of fuel transfer	Bulk fuel (BF), Canister exchange (CX)	Canister exchange would require standardization of tank sizes and connecting hardware.

Liquid hydrogen is much less dense than liquid oxygen, so the hydrogen fuel tank is much larger than the tank for the liquid oxygen. To be more precise, the mass of liquid hydrogen for a given rocket would fill up about 2.6 oxygen tanks of the same rocket.

This suggests the possibility of standardizing the propellant tanks based on the volume of the LO2 tank, and using 3 such tanks (for liquid hydrogen) for each liquid oxygen tank. But increasing the mass of LH2 means the value of the O/F ratio would decrease and should be evaluated. Doing the math, O/F ratio = (x kg)(1191.6 kg/m3) / (3x kg)(70.99 kg/m3)

= 5.596

Using liquid hydrogen and liquid oxygen at the same densities used with the Space Shuttle (70.99 kg/m^3 for liquid hydrogen and 1191.6 kg/m^3 for liquid oxygen), the resulting oxidizer-to-fuel ratio for canister tanks is about 5.596, within the acceptable range for LH2/LO2 rocket engines. (See Figure 2) Additional calculations are given in Appendix E.

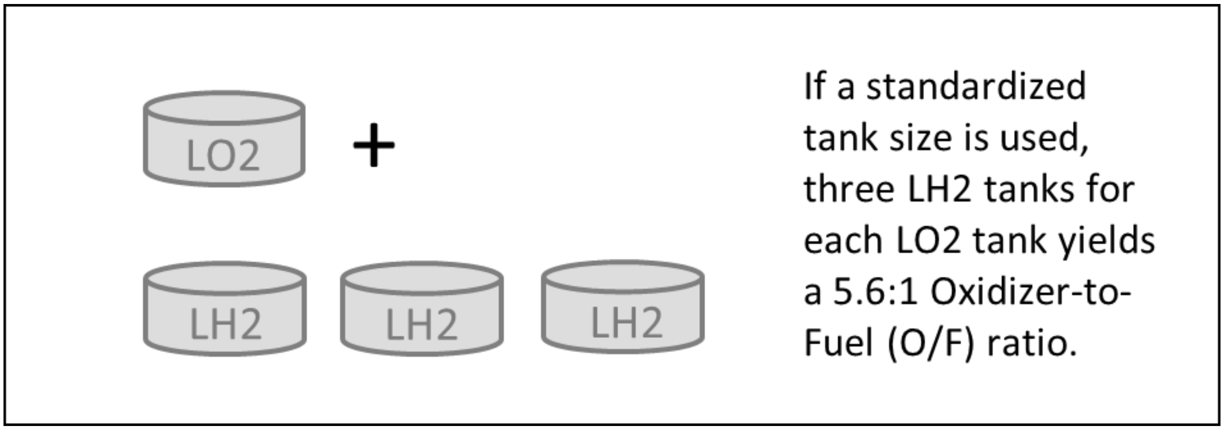

Figure 2. Oxidizer-to-Fuel Ratio Illustration

Candidate Architectures Defined

Having chosen the parameters to be examined, and the potential values or states for each parameter, it is then a simple matter to develop all of the possible combinations using those values and states.

The candidate architectures developed are shown in Table 2. In the first third of the table, electrolysis is performed in orbit on the depot. Pure water from a lunar processing facility is shipped to the depot, and the depot can be located at L1, GEO, or LEO. Once the electrolysis and liquefaction has been accomplished, the depot stores the propellants until such time as they are transferred to the customer. There are two methods of transfer – bulk fluid or canister. The second third of the table is similar, except that electrolysis and liquefaction are performed on the Moon. The propellant is then shipped to the depot location for storage and distribution.

In the bottom third of the table, electrolysis and liquefaction are performed on the Moon, but the propellants produced are also stored in a lunar depot. In this case, the propellants would

Table 2. Candidate Architectures Defined

Location of electrolysis	Location of depot	Location of transfer	Method of transfer	Remarks
In orbit	L1	L1	BF	Water is shipped from the lunar processing facility to the depot. Electrolysis and liquefaction take place on the depot.
In orbit	L1	L1	CX	
In orbit	GEO	GEO	BF	
In orbit	GEO	GEO	CX	
In orbit	LEO	LEO	BF	
In orbit	LEO	LEO	CX	
Moon	L1	L1	BF	Propellant is shipped from the lunar processing facility to the depot.
Moon	L1	L1	CX	
Moon	GEO	GEO	BF	
Moon	GEO	GEO	CX	
Moon	LEO	LEO	BF	
Moon	LEO	LEO	CX	
Moon	Moon	L1	BF	Electrolysis/fuel process-sing takes place on the Moon, and the depot is also on Moon. Tanker vehicles delivers fuel and oxidizer directly to the customer.
Moon	Moon	L1	CX	
Moon	Moon	GEO	BF	
Moon	Moon	GEO	CX	
Moon	Moon	LEO	BF	
Moon	Moon	LEO	CX	

be delivered directly from the Moon to the customer. Performing these functions on the Moon would likely take advantage of the abundant, but oblique, sunshine at the lunar poles.

Initial Architecture Network Diagram

Each of the candidate architectures shown in Table 2 represents a series of choices made – where to perform the electrolysis, where to locate a depot, where to transfer the propellants, and how to transfer the propellants. It is possible to depict these candidate architectures as separate paths from the Moon (the source of the propellants) to the final customers (J. Casler, personal communication, March 3, 2014). Assembling the many paths together forms a network diagram, with the Moon shown at the left shown in Figure 3. Each architecture is represented by a unique path through the network. The complete network diagram -- with customers -- will be presented later.

Node 1 is the processing facility on the Moon where excavated ice is melted and filtered and otherwise purified. Segments 1-2 and 1-3 represent the shipment of purified water to be loaded on tanker vehicles for transport to a depot for electrolysis and liquefaction. Segments 1-4 and 1-5 represent the shipment of LH2/LO2 to tankers to be delivered to an orbiting depot or delivered directly to customer vehicles. Nodes 6-11 represent depot locations where electrolysis and liquefaction are performed on the depot. Nodes 12-17 represent propellant delivery to a depot. Nodes 18-23 represent direct delivery of propellant to the customer vehicle(s).

As shown on the diagram, each segment shown involves the movement of fluid (either water or propellants) in the network and involves propellant consumption and losses. These are calculated in the thesis effort and are presented later. Recognize also that each candidate architecture can be described by the sequence of nodes, i.e., 1-2-6, 1-3-10, and so forth.

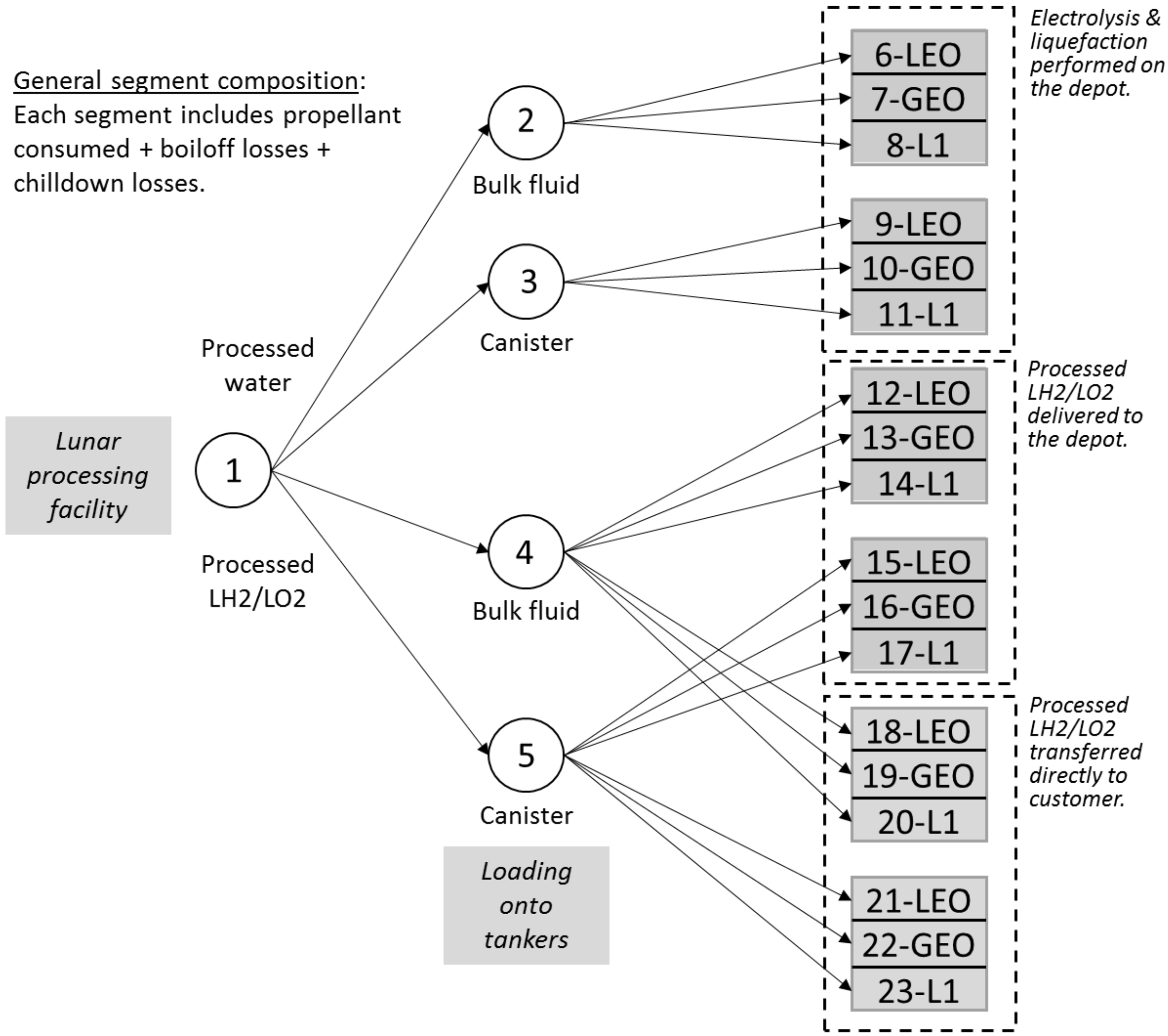

Figure 3. Initial Architecture Network Diagram

As noted on the diagram, each segment includes propellant consumed, propellant losses due to boiloff, and chilldown losses. Chilldown losses are incurred when transferring cryogenic propellant from one container to another. Propellant is intentionally drained into the transfer pipe and allowed to boil off, thus cooling the pipe and preventing further losses during the transfer.

For this study, it is assumed that tanker vehicles depart the Moon with full tanks. That is, the study assumes no consumption or losses for the initial segments 1-2, 1-3, 1-4, and 1-5.

CHAPTER IV

DESIGN REFERENCE MISSIONS

Design reference missions (DRM) are necessary to complete the candidate architectures and model the consumption and loss of propellant. Three DRMs are created. The first is a commercial satellite servicing (CSS) mission. The second is a Government Mars Cargo mission. Each of these design reference missions requires propellants to accomplish its tasks, and thus creates a demand on the architecture. A third, the Propellant Delivery Mission, is created to supply the demand by transporting fuel, or water, to the depot. Each of these is described below.

Commercial Satellite Servicing Mission and Vehicle

The amount of hydrazine on board commercial satellites limits the useful life of the satellite (Oeftering, 2011). After the hydrazine is expended, the satellite is no longer able to alter its orbit or perform station-keeping. Many satellites must then be abandoned, and their high orbits (often in GEO) make them virtually inaccessible to a manned repair or salvage mission.

This is the value of the Commercial Satellite Servicing Mission. Based at the International Space Station (ISS), the CSS mission uses an in-space vehicle (the Commercial Satellite Servicing Vehicle, or CSSV) with a LH2/LO2 engine. The vehicle carries a robotic payload, spare parts, tools, and hydrazine. Each month, the vehicle undocks from the ISS, achieves geostationary orbit, and rendezvous with and repairs or services ten satellites. The CSSV then flies to the depot and refuels, and returns to the ISS to receive new supplies and expendables, and then waits for the next mission. This DRM assumes one sortie every month.

The dry mass of the vehicle is 4,000 kilograms. The vehicle carries a robotic satellite servicer (500 kg) and carries 2,000 kg of hydrazine. The CSSV transfers 200 kg of hydrazine to each satellite serviced.

The CSSV is powered by a single Aerojet Rocketdyne model RL10B-2 rocket engine. The engine has a specific impulse (vacuum) of 465.5 seconds and generates 24,750 pounds of thrust (Aerojet Rocketdyne, 2015). A summary of the commercial satellite servicing mission and vehicle are given in Figure 4.

Commercial Satellite Servicing Mission and Vehicle (CSSV)

- Notional in-space vehicle; docked at the ISS; periodically resupplied with parts and hydrazine for satellite servicing.
- Departs ISS, achieves GEO orbit, rendezvous with and services satellites, travels to and refuels at the depot, and returns to the ISS. Ten satellites per mission; one mission per month.
- Dry mass: 4000 kg; Payload (robotic servicer): 500kg; Additional payload: 2000 kg N2H4 (hydrazine); propellant capacity varies by depot location.
- Powered by single RL10B-2 engine; I_{sp} = 465.5 sec
- Sizing based on available data for the proposed MacDonald-Dettwiler & Associates satellite servicer (Space Infrastructure Servicing, 2015).

Figure 4. Commercial Satellite Servicing Mission and Vehicle (CSSV)

Mars Cargo Mission and Vehicle

The second DRM is a Government cargo mission to Mars. This DRM is adapted from NASA’s Exploration Systems Architecture Study (ESAS) (NASA, 2005). The ESAS laid out NASA’s plans for going back to the Moon and on to Mars. For the Mars mission, NASA planned to send four cargo vehicles to Mars which would arrive there in advance of the astronaut crew.

The four vehicles would carry supplies, a Mars habitat, rovers, and anything else needed. Once the cargo vehicles had arrived safely, the astronauts would then follow in a separate crew vehicle.

The cargo vehicles themselves were the upper stage (Earth Departure Stage, or EDS) launched as a part of a heavy lift Ares V vehicle. Four Ares V rockets with EDS were to be launched over a period of 26 months (NASA, 2005, p.10). (Figure 5)

Government Mars Cargo Mission & Vehicle (MCV)

- Heavy lift vehicle to pre-position equipment and supplies in Mars orbit prior to crew arrival; based on Ares V Earth Departure Stage (EDS).
- The mission assumes four vehicles; one vehicle launch every 6 months.
- The MCV is initially launched into 200 km orbit. It docks with its cargo payload, rendezvous with the depot and refuels, and performs TMI.
- Dry mass: 24,000 kg; Payload mass: 38,600 – 52,000 kg, depending on depot location.
- Powered by single J2-X engine; I_{sp} = 449 sec
- Max fuel mass of 250,000 kg; 103,350 kg remains after initial launch into LEO.

Figure 5. Government Mars Cargo Mission and Vehicle (MCV)

In the ESAS study, the EDS vehicles were assumed to be powered by nuclear-thermal propulsion (NTP). Nuclear thermal propulsion has two advantages over chemical propulsion. It has a specific impulse (I_{sp}) roughly double that of the best chemical engines – as much as 925 seconds -- yet overall much less mass. For the purposes of this study, however, the EDS configured for the ESAS missions to the Moon is used instead. This EDS is powered by a single LH2/LO2 J-2X engine with an I_{sp} (vacuum) of 449 seconds. (Kyle, 2010, p.3)

In the ESAS study, the heavy lift vehicle places the EDS and its payload into a 200 km/ 28.5 degree orbit (S. Cook, 2008). The EDS docks with a lunar lander, and performs a trans-lunar injection from LEO. For the Mars Cargo DRM, the MCV is delivered to the same orbit as the EDS. The MCV docks with its cargo, maneuvers to the depot and refuels. Refueling at the depot enables the MCV to perform the trans-Mars injection (TMI) maneuver and the Mars Orbit Insertion upon arrival. Like the EDS, the MCV launches with 250,000 kg of propellant. After achieving LEO, the MCV has 103,500 kg of propellant remaining (Kyle, 2010).

Propellant Delivery Mission and Vehicle

The Propellant Delivery Mission satisfies the need to deliver fuel and oxidizer (or water) to the depot, or directly to the CSSV or MCV. The mission is built around a fleet of Lunar Tanker Vehicles (LTV). (Note that no attempt is made to determine an optimal number of LTVs. However, consideration should be given to at least two, in case one is down for repairs.) The LTV is an unmanned vehicle. Like the Mars Cargo Vehicle, it is powered by a J-2X engine, but has slightly less mass. Unlike the MCV, the LTV is a true in-space vehicle, and does not have to contend with an atmosphere. Consequently, the LTV is imagined as a rigid truss upon which necessary components (engine, fuel tanks, and so forth) are attached. The LTV is loaded with fuel, or water, and delivers its payload to the depot, or perhaps the other vehicles directly.

A thrust-to-weight ratio of 3 was used in determining maximum lift capacity of the LTV. (See Appendix F -- LTV Thrust-to-Weight Calculation for those calculations.) Mission designers at NASA's Marshall Space Flight Center typically express thrust-to-weight ratio (T/W_0) for lunar or planetary landers in terms of Earth's gravity, and advise that, "Optimal vehicle T/W_0 for both lunar descent & lunar ascent just happen to be at ~0.5 Earth g's." (L. Kos, personal communication, August 11, 2015) This rule of thumb is documented in greater detail by Sostaric and Merriam in their 2008 paper *Lunar Ascent and Rendezvous Trajectory Design*:

> The minimum Δv point occurs around a T/W=0.6. Increases in T/W cause a slight increase in Δv. Decreases in T/W can become quite costly [in terms of increased Δv], particularly below T/W=0.4. *The optimum point to minimize overall vehicle mass tends to be less than the minimum delta-V point, due to propulsion and structural considerations.* (emphasis added) (Sostaric and Merriam, 2008, p.9)

The characteristics of the LTV are summarized in Figure 6.

Propellant Delivery Mission & Vehicle (LTV)

- Notional lunar tanker vehicle (LTV).
- Transports LO2/LH2 or water to a fuel depot, or LO2/LH2 directly to customer vehicles (CSSV or MCV), and returns to the Moon.
- Flies as often as necessary to support demand.
- Dry mass: 22,470 kg (20,000 kg structure + 2,470 kg engine)
- Powered by single J-2X engine; I_{sp} = 449 s; Thrust = 1,307 kN (294,000 lbf)
- Using a 3:1 thrust-to-weight ratio, max takeoff weight = 268,071 kg*

(* Fuel needed by the LTV to fly to and from the depot or customer location by definition reduces the allowable payload mass.)

Figure 6. Propellant Delivery Mission and Vehicle (LTV)

Complete Architecture Network Diagram

With the addition of the CSSV and MCV, the network diagram first illustrated in Figure 3 can now be expanded (Figure 7). The figure shows the “supply side”, i.e., flights from the Moon using the LTV on the left as before, and shows delivery to the depot or customer vehicles

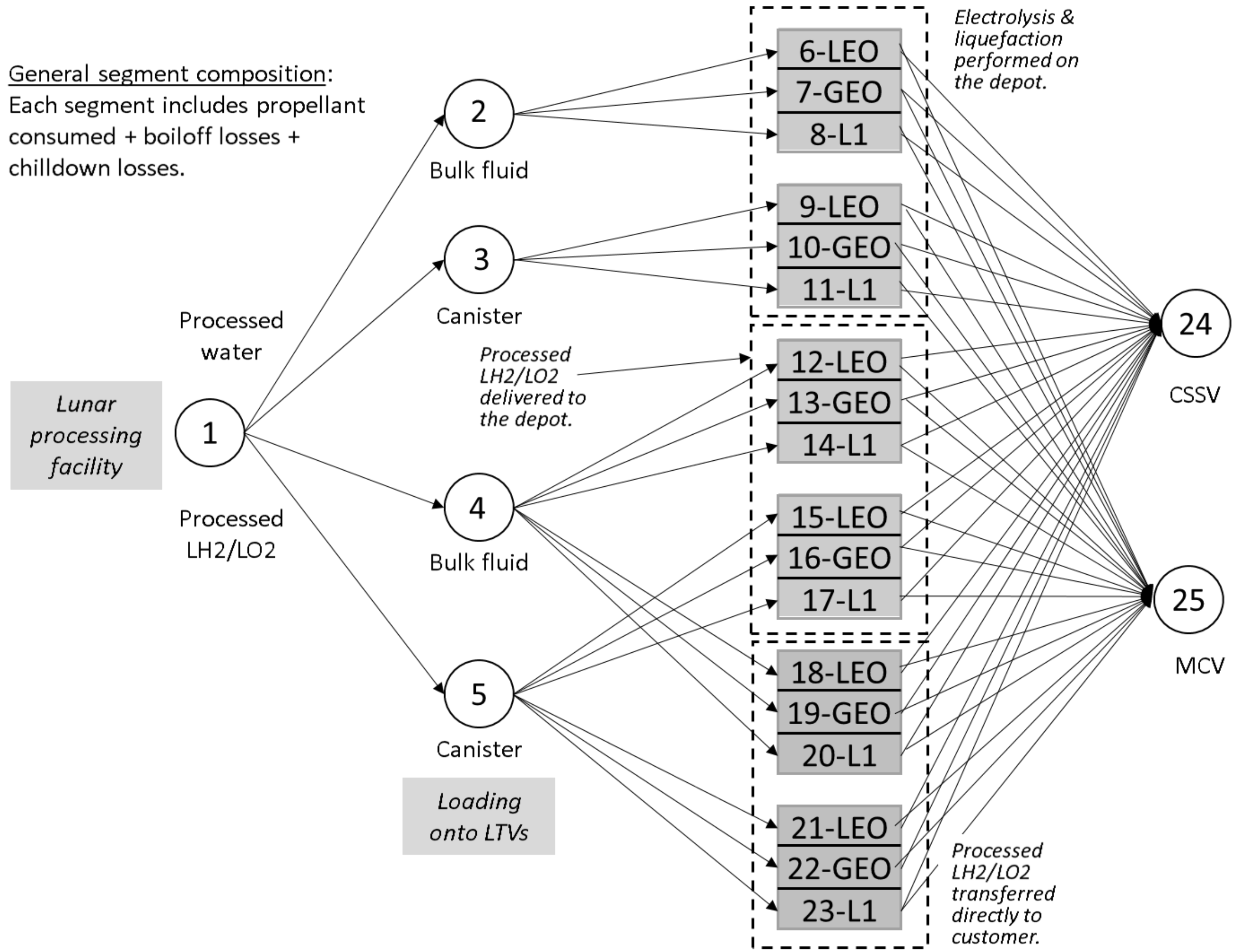

Figure 7. Complete Architecture Network Diagram

in the center of the diagram. The nodes representing the CSSV and MCV have been added to the right hand side of the diagram.

Network Diagram Segment Activities

The next step in the methodology is to calculate the Δv and time-of-flight information as a necessary step to determining fuel consumption and fuel loss due to boiloff. But to do so, it is first necessary to examine the network diagram and describe those activities taking place in each diagram segment.

The activities within each network diagram segment are listed in Table 3. Each segment is named for the node that precedes it and the node that follows it. The activities that take place in segment 2-6 are the activities that take place between nodes 2 and 6. In Table 3, the left hand column lists the many segments, while the right hand column lists the activities that take place in that segment, in terms of the architecture defining parameters – deliver fuel in bulk, deliver fuel in canisters, and so forth.

Table 3. Network Diagram Segment Activities

Segment	Segment Activity
1 – 2	Processed water loaded onto tanker vehicle as bulk fluid.
1 – 3	Processed water loaded onto tanker vehicle in canisters.
1 – 4	LH2/LO2 loaded onto tankers as bulk fluids.
1 – 5	LH2/LO2 loaded onto tankers in canisters.
2 – 6	Tankers deliver bulk water to depot in LEO.
2 – 7	Tankers deliver bulk water to depot in GEO.
2 – 8	Tankers deliver bulk water to depot in L1.
3 – 9	Tankers deliver water in canisters to depot in LEO.
3 – 10	Tankers deliver water in canisters to depot in GEO.

Table 3. cont.

Segment	Segment Activity
3 -- 11	Tankers deliver water in canisters to depot in L1.
4 – 12	Tankers deliver bulk LH2/LO2 to depot in LEO.
4 – 13	Tankers deliver bulk LH2/LO2 to depot in GEO.
4 – 14	Tankers deliver bulk LH2/LO2 to depot in L1.
4 – 18	Tankers deliver bulk LH2/LO2 to customer vehicle in LEO.
4 – 19	Tankers deliver bulk LH2/LO2 to customer vehicle in GEO.
4 – 20	Tankers deliver bulk LH2/LO2 to customer vehicle in L1.
5 – 15	Tankers deliver LH2/LO2 in canisters to depot in LEO.
5 – 16	Tankers deliver LH2/LO2 in canisters to depot in GEO.
5 – 17	Tankers deliver LH2/LO2 in canisters to depot in L1.
5 – 21	Tankers deliver LH2/LO2 in canisters to customer vehicle in LEO.
5 – 22	Tankers deliver LH2/LO2 in canisters to customer vehicle in GEO.
5 – 23	Tankers deliver LH2/LO2 in canisters to customer vehicle in L1.
6 – 24	CSS vehicle receives bulk LH2/LO2 from depot in LEO.
6 – 25	Mars cargo vehicle receives bulk LH2/LO2 from depot in LEO.
7 – 24	CSS vehicle receives bulk LH2/LO2 from depot in GEO.
7 – 25	Mars cargo vehicle receives bulk LH2/LO2 from depot in GEO.
8 – 24	CSS vehicle receives bulk LH2/LO2 from depot in L1.
8 – 25	Mars cargo vehicle receives bulk LH2/LO2 from depot in L1.
9 – 24	CSS vehicle receives LH2/LO2 in canisters from depot in LEO.
9 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from depot in LEO.
10 – 24	CSS vehicle receives LH2/LO2 in canisters from depot in GEO.
10 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from depot in GEO.
11 – 24	CSS vehicle receives LH2/LO2 in canisters from depot in LEO.
11 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from depot in L1.
12 – 24	CSS vehicle receives bulk LH2/LO2 from depot in LEO.
12 – 25	Mars cargo vehicle receives bulk LH2/LO2 from depot in LEO.
13 – 24	CSS vehicle receives bulk LH2/LO2 from depot in GEO.
13 – 25	Mars cargo vehicle receives bulk LH2/LO2 from depot in GEO.

Table 3. cont.

Segment	Segment Activity
14 – 24	CSS vehicle receives bulk LH2/LO2 from depot in L1.
14 – 25	Mars cargo vehicle receives bulk LH2/LO2 from depot in L1.
15 – 24	CSS vehicle receives LH2/LO2 in canisters from depot in LEO.
15 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from depot in LEO.
16 – 24	CSS vehicle receives LH2/LO2 in canisters from depot in GEO.
16 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from depot in GEO.
17 – 24	CSS vehicle receives LH2/LO2 in canisters from depot in L1.
17 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from depot in L1.
18 – 24	CSS vehicle receives bulk LH2/LO2 from tanker in LEO.
18 – 25	Mars cargo vehicle receives bulk LH2/LO2 from tanker in LEO.
19 – 24	CSS vehicle receives bulk LH2/LO2 from tanker in GEO.
19 – 25	Mars cargo vehicle receives bulk LH2/LO2 from tanker in GEO.
20 – 24	CSS vehicle receives bulk LH2/LO2 from tanker in L1.
20 – 25	Mars cargo vehicle receives bulk LH2/LO2 from tanker in L1.
21 – 24	CSS vehicle receives LH2/LO2 in canisters from tanker in LEO.
21 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from tanker in LEO.
22 – 24	CSS vehicle receives LH2/LO2 in canisters from tanker in GEO.
22 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from tanker in GEO
23 – 24	CSS vehicle receives LH2/LO2 in canisters from tanker in L1.
23 – 25	Mars cargo vehicle receives LH2/LO2 in canisters from tanker in L1.

Objective Function for the Study

In examining Table 3, the reader will notice the verbs “delivers” and “receives” are used repeatedly. These words describe the activities only in the most general sense. Delivering or receiving propellants involves a number of separate actions. A more detailed explanation of the separate actions in the network segments is worthwhile, and assists in developing the Objective Function for this study.

Consider segment 4-13 “Tankers deliver bulk LH2/LO2 to depot in GEO.” This sounds simple, but is not. Several steps are involved, and each has its own cost in terms of propellant. In this case, to deliver bulk propellants to the depot in GEO, the LTV must launch from the Moon and maneuver to the depot. Travel to the depot requires the consumption of the LTV’s propellant, which must be calculated. But the LTV’s propellant tanks are also subject to propellant boiloff during the journey. This boiloff, too, must be calculated. In addition, the LTV is carrying propellants to deliver to the depot. The propellants themselves are subject to boiloff during the journey to the depot. After arriving at the depot, the propellants must be transferred from the LTV to the depot. The pipes that carry the propellants from the LTV to the depot must be chilled before the transfer can take place. This process is called “chilldown”, and causes some propellant to be lost. Finally, the LTV must fly back to the Moon, again with the attendant consumption and loss of its own propellants.

A similar story can be told with respect to the customer. Consider segment 13-24 “CSS vehicle receives bulk LH2/LO2 from depot in GEO.” This segment is quite complicated. For this segment, the CSSV departs its home base at the International Space Station (ISS), maneuvers from the ISS’ orbit to geostationary orbit. After doing so, it maneuvers to each of 10 satellites in GEO orbit and services them. It then maneuvers to the depot and refuels, then returns to the ISS. Each of its maneuvers requires the consumption of propellant, and during the entire time (including time spent at the ISS), it is subject to losing propellant to boiloff. And as with the transfer of propellants from the LTV to the depot, the transfer of propellants from the depot to the CSSV also includes a chilldown loss.

Thus, it can be seen that every activity in a given architecture can be measured in terms of its propellant consumption or propellant loss due to boiloff. This allows the objective function to then be formulated. Recall the research question - "Which architecture satisfies the Design Reference Missions (DRMs) for the least amount of liquid oxygen (LO2) and liquid hydrogen (LH2) consumed in flight or lost due to boiloff?" We are seeking to identify the architecture with the least overall consumption and loss of propellant:

Objective Function: Minimize: $X_{ijk} = P_{LTV} + B_{LTV} + B_{P/L} + C_{P/L} + P_{CSSV} + C_{CSSV} + B_{CSSV} + P_{MCV} + C_{MCV} + B_{MCV}$

where X_{ijk} maps to a unique candidate architecture (unique path in the network diagram), and

P_{LTV} = Propellant consumed by the LTV

B_{LTV} = Boiloff losses of the LTV's own propellant

$B_{P/L}$ = Boiloff losses for the LTV payload

$C_{P/L}$ = Chilldown losses when transferring the LTV payload to the depot

P_{CSSV} = Propellant consumed by the CSSV

C_{CSSV} = Chilldown losses when the CSSV receives propellants

B_{CSSV} = Boiloff losses on the CSSV

P_{MCV} = Propellant consumed by the MCV

C_{MCV} = Chilldown losses when the MCV receives propellants

B_{MCV} = Boiloff losses on the MCV

Recall in the descriptions of the design reference missions that the Commercial Satellite Servicing Vehicle (CSSV) "flies" every month, while the Mars Cargo Vehicle (MCV) launches once every six months. In order to properly compare the candidate architectures using the objective function, the values calculated will be for a six month period – i.e., six missions by the

CSSV and one mission by the MCV. The number of supply missions flown by the LTV(s) will be based on the need for propellants. Since each candidate architecture will supply both the CSSV and MCV, each architecture will be designated by only three numbers, such as 1-3-10 or 1-4-19. This will be done for convenience. Doing so is simpler than describing architecture pairs, such as 1-3-10-24 and 1-3-10-25 or a combined designation 1-3-10-24/25. The delivery to the CSSV and MCV (nodes 24 and 25) is understood.

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CHAPTER V

ORBITAL MECHANICS

Assumptions Used

To simplify calculations, restricted two-body techniques were used for all calculations. Instantaneous accelerations were assumed. Circular, coplanar orbits were assumed for the Earth, Moon, Mars, and the depot in orbit. Likewise, these orbits were assumed to be coplanar with the Sun. For the CSSV performing co-orbital rendezvous to service satellites in GEO, the ten customer satellites were assumed to be evenly distributed 36 degrees apart. For the CSSV or MCV performing a co-orbital rendezvous with a depot in GEO, the depot was assumed to be 180 degrees ahead (i.e. worst case). The LEO orbit for the depot was assumed to be 400 km altitude, 0 degrees inclination.

Delta-v and Time-of-Flight Values for Individual Maneuvers

As described in the previous chapter, each segment on the network diagram represents a lot of activity, whether the LTV in delivering propellants, the CSSV servicing satellites and then going to refuel, or the MCV docking with its cargo payload and maneuvering to the depot to refuel before executing trans-Mars-injection (TMI). A stepping stone approach was used, in that the Δv and time-of-flight values were first calculated for each individual maneuver, such as “ISS to GEO”, or “rendezvous in GEO”, and so forth. These Δv and time-of-flight values were then combined to produce values for mission Δv and time-of-flight. The Δv and time of flight for individual maneuvers is given in Table 4 below.

Table 4. Delta-v and Time-of-Flight Values for Individual Maneuvers

Maneuver (Application)	Δv (km/s)	Time-of-Flight (hrs)
Depot in LEO (400/0)		
GEO-to-LEO (400/0)) (CSSV rendezvous with depot in LEO)	3.854	5.3
Co-orbital rendezvous in LEO (400/0) (LTV rendezvous with depot; depot 180^{o} ahead)	2.031	2.3
LEO (400/0) to ISS CSSV return from depot at LEO to ISS	6.683	0.8
LEO (200/28.5) to LEO (400/0) MCV from initial orbit to depot	3.889	0.8
LEO (400/0) -to-Mars (MCV departing LEO enroute to Mars)	5.670	288 days
Moon-to-LEO (400/0) LTV deliver to depot at LEO	6.287	119.6
LEO (400/0)-to-Moon LTV return to Moon from LEO	6.287	119.6
Depot in GEO		
ISS-to-GEO (CSSV to GEO to service satellites)	4.838	5.3
Co-orbital rendezvous in GEO (CSSV rendezvous with customer satellite)	0.232	21.5
Co-orbital rendezvous in GEO (CSSV rendezvous with depot; depot 180^{o} ahead)	1.905	12.0
GEO-to-ISS (CSSV return from GEO to ISS)	4.839	5.3
LEO (200/28.5)-to-GEO (MCV goes to GEO to refuel at depot there)	4.291	5.3
GEO-to-Mars (MCV departing GEO enroute to Mars)	4.278	288 days
Moon-to-GEO (LTV delivers fuel to GEO)	3.995	136.1
GEO-to-Moon (LTV returning to Moon after delivery to GEO)	3.995	136.1

Table 4. cont.

Maneuver (Application)	Δv (km/s)	Time-of-Flight (hrs)
Depot at L1		
GEO-to-L1 (CSSV going to L1 to refuel)	1.332	107.4
L1-to-ISS (CSSV returning to ISS after refueling at L1)	3.811	92.2
LEO (200/28.5)-to-L1 (MCV going to L1 to refuel)	3.780	92.1
L1-to-Mars (MCV departing L1 enroute to Mars)	4.327	288 days
Moon-to-L1 (LTV delivering fuel to L1)	2.342	65.6
L1-to-Moon (LTV returns to Moon from L1)	2.342	65.6

After the calculations for the Δv and time of flight for individual maneuvers was completed, the next task was to group the values to understand the Δv and time of flight for entire missions for each of the design reference missions.

The time of flight for travel to Mars was based on "conjunction class" trajectories where the Earth at launch and Mars at arrival are nearly in direct opposition. Nine such launch opportunities from the year 2002-2011 are recorded in the NASA's Interplanetary Mission Design Handbook. (George & Kos, 1998, p.136) The Δv and time of flight values were averaged. The average Δv was 3.673 km/sec, which compares favorably to the "Δv boost" value of 3.569 km/sec calculated using the patched conic method. The average time of flight over the nine flights was 288 days. This value was used in subsequent boiloff calculations.

Δv and time-of-flight tables (Tables 5, 6, and 7) for the CSSV, MCV, and LTV are provided on the pages that follow.

Table 5. Delta-v and Time-of-Flight Values – CSSV

Design Reference Mission (DRM)	CSS Vehicle Maneuver	Δv (km/s)	TOF (hrs)	Remarks
Commercial Satellite Servicing Vehicle (CSSV)				
Depot or LTV in LEO	ISS – to - GEO	4.838	5.3	HT w/plane change at apogee
	Rendezvous w/10 satellites in GEO	2.320	215.4	Co-orbital rendezvous
	Return to LEO (400/0) to refuel	3.854	5.3	HT w/plane change at apogee; assumption is made that the maneuver to LEO can be timed well enough to minimize any Δv requirements for rendezvous.
	Return to ISS (LEO 400/0 to ISS)	6.683	0.8	HT w/plane change at apogee
Totals:		17.695	226.7	
Depot or LTV in GEO	ISS – to - GEO	4.838	5.3	HT w/plane change at apogee
	Rendezvous w/10 satellites in GEO	2.320	215.4	Co-orbital rendezvous
	Rendezvous with depot in GEO	1.905	12.0	Co-orbital rendezvous; assumes depot is 180 degrees ahead (worst case)
	GEO-to-ISS	4.839	5.3	HT w/plane change at apogee
Totals:		13.902	237.9	
Depot or LTV at L1	ISS – to - GEO	4.838	5.3	HT w/plane change at apogee
	Rendezvous w/10 satellites in GEO	2.320	215.4	Co-orbital rendezvous
	Performs TLI to L1 (GEO to L1)	1.332	107.5	Completed using mean motion calculations.
	Performs TEI to LEO (L1 to ISS)	3.811	92.2	HT w/plane change at apogee
Totals:		12.301	420.3	

Table 6. Delta-v and Time-of-Flight Values – MCV

Design Reference Mission (DRM)	Mars Vehicle Maneuver	Δv (km/s)	TOF (hrs)	Remarks
Cargo Mission to Mars				
Depot in LEO or tanker delivers to LEO	Maneuver to depot in LEO (200/28.5 to 400/0)	3.889	0.8	Plane change
	Perform TMI (LEO 400/0-to-Mars)	3.569	288 days	Patched conic method
	Perform final burn to enter Martian orbit	2.101		Hyperbolic Mars arrival.
Totals:		9.559	---	
Depot in GEO or tanker delivers to GEO	LEO 200/28.5– to -GEO	4.291	5.3	HT with plane change; assumption is made that the maneuver to GEO can be timed well enough to minimize any Δv for depot rendezvous.
	Perform TMI (GEO-to-Mars)	2.177	288 days	Patched conic method
	Perform final burn to enter Martian orbit	2.101		Hyperbolic Mars arrival.
Totals:		8.569	5.3	
Depot at L1 or tanker delivers to L1	Performs TLI to L1 from LEO 200/28.5	3.780	92.1	Completed using mean motion calculations
	Rendezvous with depot	---	---	Some Δv will be expended to rendezvous/ maintain the position at L1. This is assumed to be small enough as to not affect the overall calculations.
	Performs TMI	2.226	288 days	Patched conic method
	Perform final burn to enter Martian orbit	2.101		Hyperbolic Mars arrival.
Totals:		8.107	92.1	

Table 7. Delta-v and Time-of-Flight Values – LTV

Design Reference Mission (DRM)	Lunar Tanker Maneuver	Δv (km/s)	TOF (hrs)	Remarks
Lunar Tanker				
Delivers fuel to depot or customer in L1	Launches from Moon – direct ascent to L1	2.342	65.6	Calculated as a rectilinear orbit from Moon
	L1 position maintenance	---	---	Some Δv will be expended to maintain the position at L1. However, this is assumed to be small enough as to not affect the overall calculations.
	Maneuvers from L1 back to lunar surface	2.342	65.6	Calculated as a rectilinear orbit to the Moon
Totals:		4.684	131.3	
Delivers fuel to depot or customer in GEO	Launches from Moon to GEO	3.995	136.1	Hohmann Transfer
	Co-orbital rendezvous in GEO	1.905	12.0	Co-orbital rendezvous
	Maneuvers from GEO back to Moon	3.999	136.1	Hohmann Transfer
Totals:		9.899	284.3	
Delivers fuel to depot in LEO (400/0)	Launches from Moon to LEO 400/0	6.287	120.0	Hohmann Transfer (HT) w/plane change at apogee
	Co-orbital rendezvous in LEO	2.031	2.3	Co-orbital rendezvous
	Maneuvers from LEO back to Moon	6.287	120.0	Hohmann Transfer (HT) w/plane change at apogee
Totals:		14.605	241.5	

The Δv and time-of-flight values for all vehicles for all potential [orbital] depot locations are summarized in Table 8.

Table 8. Summary of Mission Delta-v and Time-of-Flight Values

Mission	Δv (km/s)	Time of Flight (hrs)
CSSV departs ISS, services satellites, maneuvers to depot, refuels, and returns to ISS.		
Depot in LEO (400/0)	17.695	226.7
Depot in GEO	13.902	237.9
Depot in L1	12.301	420.3
MCV docks w/cargo in LEO parking orbit, maneuvers to depot, refuels, and departs for Mars.		
Depot in LEO (400/0)	9.559	24.75 + travel to Mars (288 days)[1]
Depot in GEO	8.569	29.26 + travel to Mars (288 days)[1]
Depot in L1	8.107	116.1 + travel to Mars (288 days)[1]
LTV departs Moon, travels to depot/customer, transfers fuel, and returns to Moon.		
Depot in LEO (400/0)	14.605	241.5
Depot in GEO	9.899	284.3
Depot in L1	4.684	131.3
Notes: [1]Includes 24 hours spent in LEO after launch.		

It is apparent from the values for Δv in the table that LEO is the most stressing depot location for all three DRM vehicles. Likewise, L1 is the least stressing depot location for all three DRM vehicles. But it is also noteworthy that the time-of-flight for the CSSV and MCV in cis-lunar space increases markedly when the depot is located at L1. This increased time of flight suggests these vehicles will experience greater boiloff during their missions, and there might be some tradeoff between reduced Δv requirements and increased boiloff.

CHAPTER VI

FUEL CONSUMPTION AND PROPELLANT DELIVERY

Fuel Consumption

With the Δv values established for the various maneuvers and design reference missions, the next step is to calculate the expected fuel consumption for each of the vehicles. Multiple iterations of the rocket equation were used. For each vehicle, a backwards planning approach was used. To illustrate, the simplest case among all three vehicles was for the lunar tanker vehicle (LTV) delivering fuel to the depot located at L1. That is, the first question asked was "How much fuel will be needed to fly the [empty] LTV back to the Moon after it makes its delivery?" Once that mass of fuel was determined, that mass together with the dry mass of the vehicle and the payload mass became the m_{final} used in the next iteration of the rocket equation. The next $m_{initial}$ calculated represented total vehicle mass – dry mass, payload mass, fuel to maneuver to L1, and the fuel to maneuver the empty LTV back to the Moon. All calculations were performed using spreadsheet software.

CSSV Fuel Consumption Calculations

For the Commercial Satellite Servicing Vehicle (CSSV) a similar approach was used. Recall that the CSSV starts its mission at the International Space Station. It maneuvers to GEO, and rendezvous with and services ten different satellites. It transfers 200 kg of hydrazine to each satellite. It then maneuvers to the depot and refuels, and returns to the ISS and receives a new payload (2,000 kg) of hydrazine.

The fuel consumption calculation for the CSSV begins with figuring the mass of propellant needed to maneuver the empty CSSV (CSSV dry mass plus its robotic servicer) from the last satellite serviced to the depot. As with the example above, this mass of fuel is added to the mass of the empty CSSV and becomes the m_{final} for the next iteration of the rocket equation. But there is a twist. In the backward progression through the mission, 200 kg of hydrazine must be added for each of the ten rendezvous maneuvers. So 200 kg of hydrazine is added to each final mass before calculating the next $m_{initial}$.

After iterating through each satellite, the next step is to calculate the fuel to maneuver the CSSV from the ISS to GEO where the satellites are located. The last step is to go from the depot to the ISS. If done correctly, the amount of fuel on the CSSV after refueling will permit maneuvering to the ISS and a full satellite servicing mission before having to refuel again. (Figure 8)

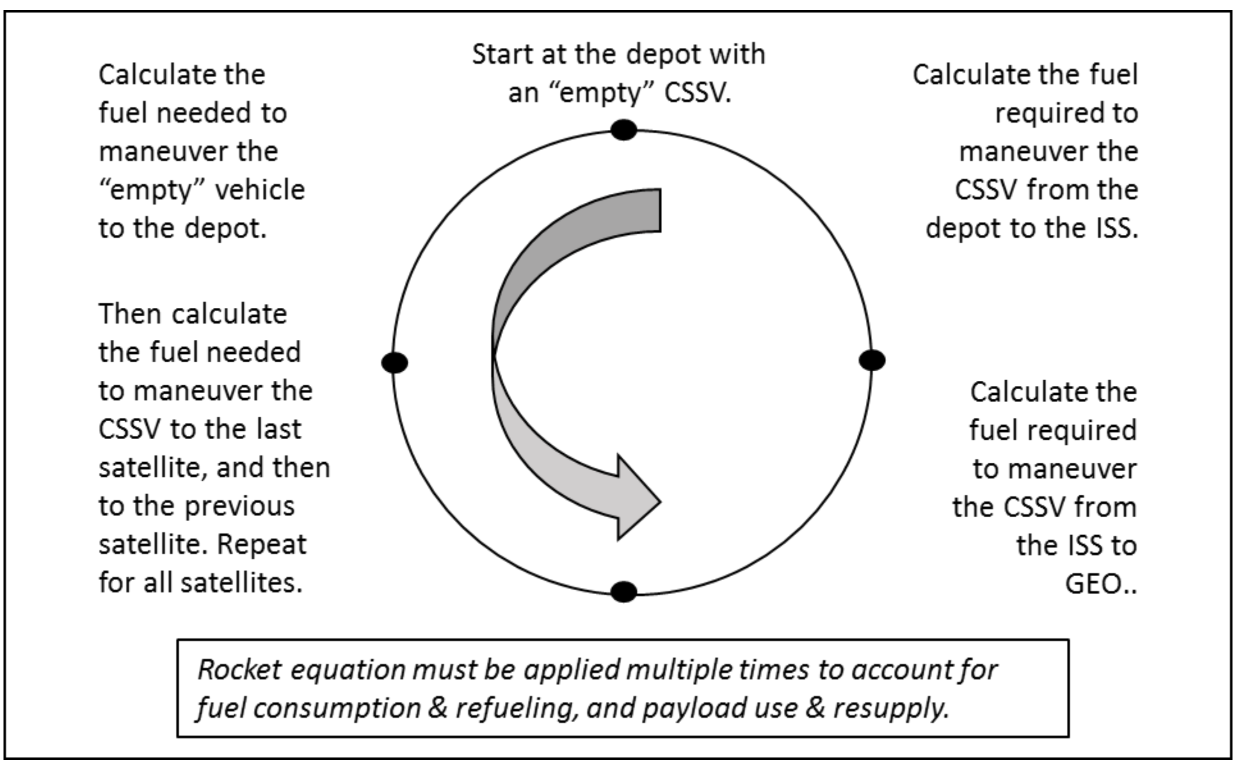

Figure 8. Fuel Calculation Method (CSSV)

MCV Fuel Consumption Calculations

The fuel consumption calculations for MCV were similar to those for the CSSV, with one exception. The MCV starts out in LEO, with 103,350 kg of propellant remaining after launch. (Kyle, 2010) So the initial step with MCV was to determine the maximum payload mass that would still allow the MCV to fly to the depot location with the remaining propellant. (After the arrival at the depot, the initial propellant from the launch is assumed to be fully consumed.) Once that payload was determined, the payload mass and the MCV dry mass become the m_{final} that must be delivered to Mars orbit. The rocket equation was then applied to calculate the propellant required for the burn to enter Mars orbit. That fuel is added to the payload mass and the MCV dry mass and becomes the next m_{final} to determine the propellant mass required for the hyperbolic departure from Earth orbit (Figure 9).

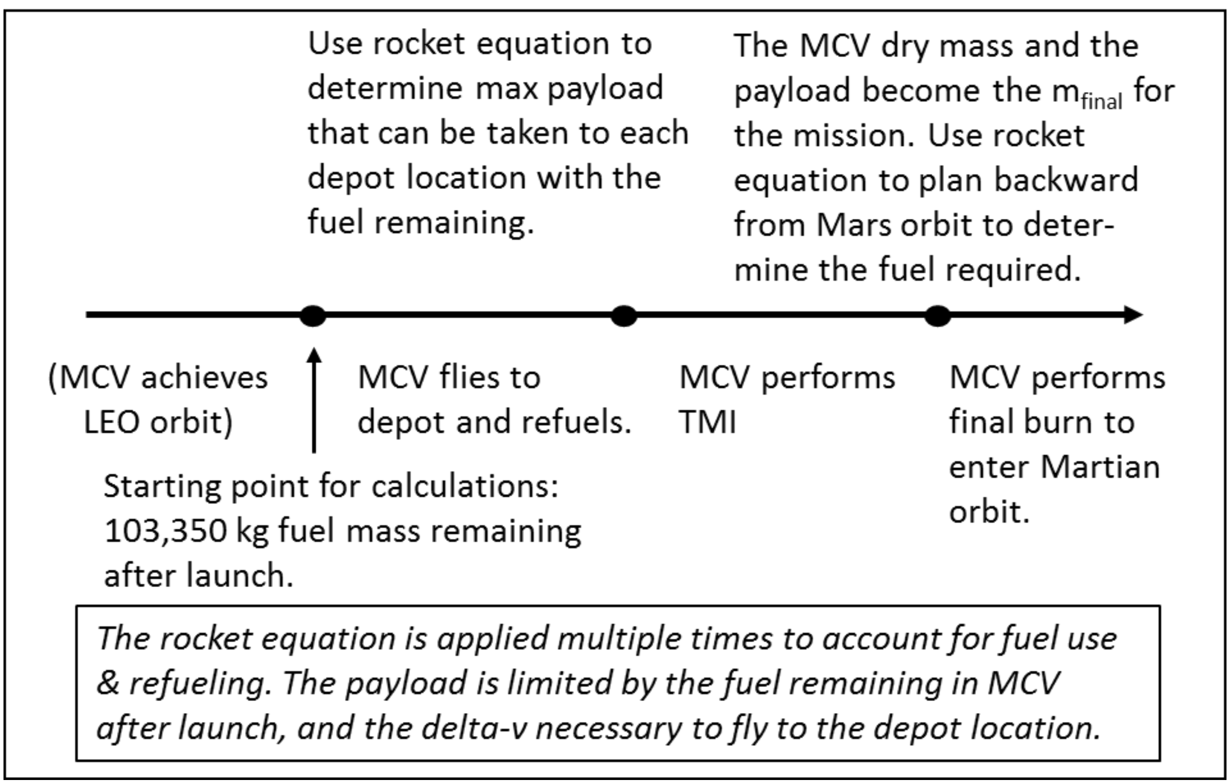

Figure 9. Fuel Calculation Method (MCV)

LTV Fuel Consumption Calculations

For the LTV calculations, the same backward planning process was used as with the other vehicles. However, for the LTV the payload is not known in advance. It follows that the amount of propellant the LTV consumes to make the trip reduces that portion of the vehicle lift capacity that can be allocated to the payload, and several iterations of the rocket equation may be needed until the right combination of fuel and payload are achieved. (Figure 10)

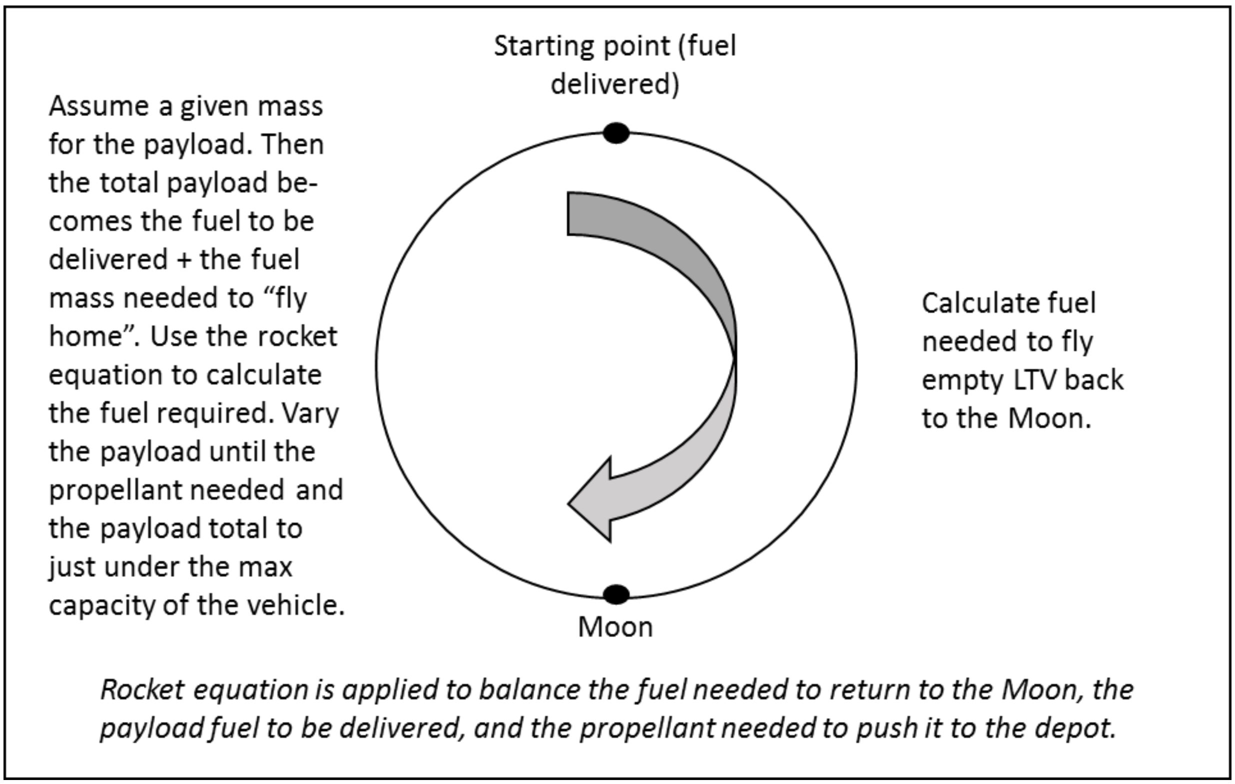

Figure 10. Fuel Calculation Method (LTV)

The results of the fuel calculations are shown in the Table 9 below. The table provides the fuel required for each vehicle, and the maximum payload for each vehicle for all three proposed depot locations. Spreadsheet calculations are included on a compact disk.

Table 9. Fuel Consumption and Maximum Payloads for CSSV, MCV, and LTV

Vehicle	Depot in LEO	Depot in GEO	Depot at L1
CSSV Fuel Required (kg) [1]	243,621	110,229	77,803
CSSV Payload (kg)	2,000	2,000	2,000
MCV Fuel Required (kg) [2]	191,075	102,740	126,978
MCV Payload (kg)	48,850	38,600	52,000
LTV Fuel Required (kg)	571,796 [3]	231,065	126,320
LTV Payload (kg)	---	14,520	119,275

Notes:

[1] CSSV fuel is that needed for one mission – departing from the ISS, servicing satellites, refueling, and returning to the ISS.

[2] MCV fuel is that needed to depart LEO and refuel at the depot, perform TMI, and have enough fuel remaining to enter Martian orbit. The fuel remaining after achieving initial LEO orbit limits the payload that can be taken forward.

[3] LTV fuel required to deliver in LEO is greater than its total lift capacity.

Propellant Delivery Calculations

The amount of fuel the Lunar Tanker Vehicle can deliver varies by the location of the depot. The more fuel the LTV needs to make the trip (and return to the Moon), the less payload mass is available for propellant that can be delivered to a depot. As described earlier, the general method is to start by calculating the amount of fuel needed to bring the empty LTV back from the depot. The example below shows some sample calculations:

Example: Calculate how much fuel the LTV can deliver to a depot in geostationary orbit (GEO).

- Dry mass of LTV is 22,470 kg
- Δv from the Moon to GEO: 3.995 km/sec
- Δv to rendezvous in GEO: 1.905 km/sec (assumes depot is 180 degrees ahead of LTV)
- Δv to fly from GEO back to the Moon: 3.999 km/sec
- I_{sp} of the LTV J-2X engine is 449 seconds; LTV max lift capability = 245,575 kg

Step 1: Calculate fuel to fly back to Moon

$\Delta v = I_{sp}\, g_0 \ln (m_i/m_f)$

3,999 m/s = (449 seconds) (9.81 m/s2) ln (m_i/22,470 kg)

Solve for m_i: $m_i = (m_f) \times e^{(\Delta v / Isp\ g0)}$ = 55,705 kg

Fuel required = $m_i - m_f$ = 55,705 – 22,470 = 33,235 kg

Step 2: Calculate fuel for rendezvous in GEO

- New m_{final} = 55,705 kg
- Δv = 1.905 km/sec

$\Delta v = I_{sp}\, g_0 \ln (m_i/m_f)$

1,905 m/s = (449 seconds) (9.81 m/s2) ln (m_i/55,705 kg)

Solve for m_i: $m_i = (m_f) \times e^{(\Delta v / Isp\ g0)}$ = 85,847 kg

Fuel required = $m_i - m_f$ = 85,847 - 55,705 = 30,142 kg

Step 3: Calculate fuel to get to the Depot

- New m_{final} = 85,847 kg
- Δv = 3.995 km/sec

$\Delta v = Isp\, g_0 \ln (m_i/m_f)$

3,995 m/s = (449 seconds) (9.81 m/s^2) ln (m_i/85,847 kg)

Solve for mi: m_i = (85,847) x e (3,995 /449 x 9.81) = 226,437 kg

Fuel required = $m_i - m_f$ = 226,437 – 85,847 = 140,590 kg

Step 4: Determine effective payload to depot in GEO.

- LTV dry mass is 22,470 kg
- LTV max lift capacity is 245,575 kg
- Fuel used in Steps 1-3: 33,235 + 30,142 + 140,590 = 203,967 kg

By subtracting the estimated fuel from the max lift capacity, this suggests the LTV could lift 245,575 – 203,967 = 41,608 kg to the depot. But since $m_{initial}$ would be greatly increased, the mass of fuel to lift it to the depot also increases, as does the mass of fuel needed for rendezvous, so the actual payload value will be less. It is then necessary to adjust the payload value, so that the mass of the payload and the mass of the fuel to deliver it, and the mass of the fuel to return to the Moon do not exceed the lift capacity of the vehicle.

Similar calculations were done for delivery to all candidate depot locations using Excel spreadsheets. These spreadsheets are in the CD provided with this thesis document.

Implications of Combined Calculations

At this point, it is instructive to examine the fuel consumption and maximum payload values to see what other information can be gleaned. Look first at the numbers for the LTV given in Table 10 below. Perhaps the most obvious fact is the mass of propellant the LTV needs for the round trip from the Moon to LEO is much greater than the lift capacity of the vehicle. This means that low Earth orbit is not a viable location for a depot supplied from lunar resources, unless some means is used to reduce the Δv requirement, as some authors have suggested.

It is also evident that propellant delivery to geostationary orbit is a poor value, where the LTV consumes much more propellant than it is able to deliver. At L1 the LTV still uses more propellant than it delivers, but almost achieves parity. Of the three orbit locations examined, L1 is clearly the most efficient location to deliver propellant.

Table 10. LTV Propellant Delivery to LEO, GEO, and L1

Depot Location	LTV Fuel Required (kg)	Quantity Fuel Delivered (kg)	Remarks
LEO	571,796	---	Amount of fuel needed for round trip exceeds capacity of LTV.
GEO	231,065	14,520	LTV uses more fuel than it delivers.
L1	126,320	119,215	LTV uses more fuel than it delivers.

The next table compares the mass the LTV can deliver to different depot locations with the propellant masses the CSSV and MCV require to perform their missions (Table 11).

Table 11. LTV Capacity to Service Design Reference Missions

Vehicle	Fuel Needed for mission (kg)	Mass LTV can deliver (kg)	Remarks
Commercial Satellite Servicing Vehicle (CSSV)			
Depot in LEO	243,621	---	LTV cannot service CSSV or depot in LEO.
Depot in GEO	110,229	14,520	LTV capacity is less than fuel required; impractical to service CSSV directly.
Depot in L1	77,803	119,275	LTV capacity is greater than fuel required; can service the depot or CSSV directly.
Mars Cargo Vehicle (MCV)			
Depot in LEO	191,075	---	LTV cannot service MCV or depot in LEO.
Depot in GEO	102,740	14,520	LTV capacity is much less than fuel required; impractical to service MCV directly.
Depot in L1	126,978	119,275	LTV capacity is less than fuel required; cannot service MCV with a single LTV.

For a depot in GEO, for example, the LTV would have to make seven trips to deliver enough fuel for the MCV. To deliver enough fuel for the CSSV, it would have to make eight trips. Notice also that if the depot is located at L1, the LTV can deliver enough fuel to service the CSSV with a single vehicle, and the MCV with two. This suggests the LTV could also service these vehicles directly, if the decision was made to locate the depot on the lunar surface.

There are two more implications of the fuel consumption figures. The first of these is for the sizing of the depot. The design reference mission for the CSSV says it will visit the depot once each month. The design reference mission for the MCV says it will launch every sixth month. This defines the depot capacity needed. The size of the depot at each location would be the sum of propellant mass needed by the CSSV and the mass needed by the MCV, since in the sixth month, both vehicles would maneuver to the depot to obtain propellant. (Table 12)

Table 12. Fuel Depot Sizing

Depot Location	CSSV Fuel Required (kg)	MCV Fuel Required (kg)	Suggested Depot Size/Remarks
LEO (400 km/0 deg)	243,621 (once per month)	191,075 (once every 6 months)	434,696 kg, based on fueling both vehicles every 6th month, but the LTV cannot service the depot in LEO.
GEO	110,229 (once per month)	102,740 (once every 6 months)	212,969 kg, based on fueling both vehicles every 6th month.
L1	77,803 (once per month)	126,978 (once every 6 months)	204,871 kg, based on fueling both vehicles every 6th month.

It can be seen that depending on the location, the suggested depot size varies considerably. If the depot in LEO were feasible, it would be more than double the size of the depot at L1.

The second implication of the fuel consumption numbers has to do with how much fuel will pass through the depot over the six month period of time used with the objective function described earlier. This quantity of fuel is called “throughput”, and it is informative to see how the values for throughput translate to the number of trips the LTV would make to service each depot location (Table 13).

Table 13. LTV Flights to Supply the Depot

Depot Location	Mass Required per Six Months (kg)	Mass LTV can deliver per flight (kg)	LTV Flights needed to service the depot
LEO[1]	1,653,401	---	---
GEO	764,114	14,520	52.625 → 53
L1	593,796	119,275	4.978 → 5[2]
Notes: [1]LTV cannot support a depot located in LEO. [2]Bulk fuel only. Canisters require 6 flights.			

This is most telling table of all. To L1, the LTV delivers almost as much as it consumes. Likewise, fuel consumption requirements for the CSSV and the MCV are the least for the depot at L1, resulting in a smaller depot and smaller throughput over a six month period. A depot at L1 would be easier to maintain, and wear and tear on the LTV fleet would be greatly reduced.

At this point, it is appropriate to update the network diagram. Since the low Earth orbit (400 km altitude, 0 degrees inclination) has been shown to not be a viable location for the fuel depot, the LEO locations on the diagram have been crossed out, and the diagram segments from the Moon to LEO and from the other DRMs to LEO have been removed (Figure 11). Since it is impractical to service the CSSV or MCV directly in GEO, diagram segments 19-24, 19-25, 22-24, and 22-25 have also been removed.

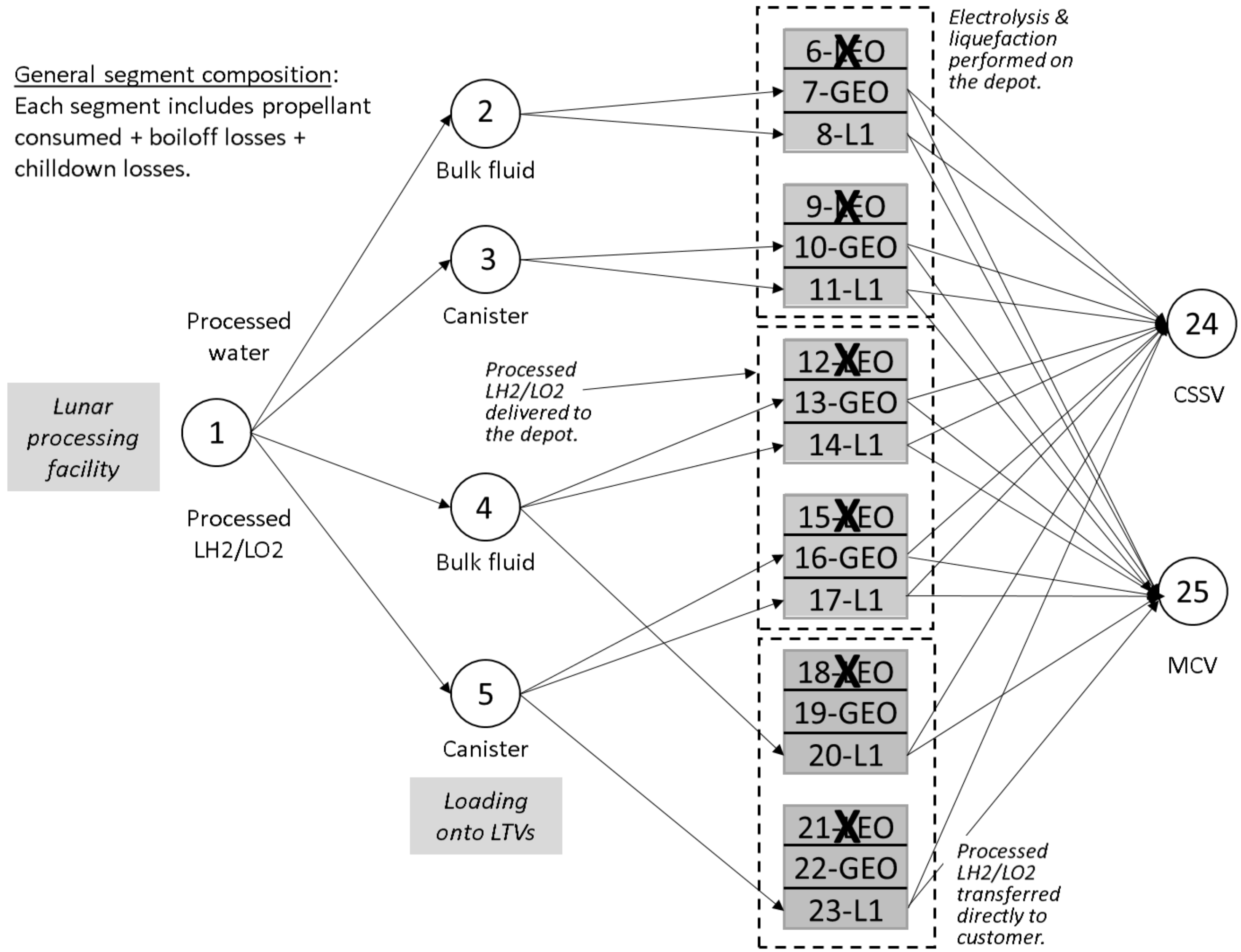

Figure 11. Updated Architecture Network Diagram

CHAPTER VII

CHARACTERIZING THE THERMAL ENVIRONMENT

Thus far, we have defined the candidate architectures for the fuel depot problem, and have defined the design reference missions that supply the depot and create demand for the depot. We have determined the Δv and time-of-flight values for the many maneuvers that would be involved with each architecture, and have calculated the propellant requirements for each architecture. The next major step will be to calculate the anticipated propellant losses due to boiloff. But to do that, we must first characterize the thermal environment in which the spacecraft operate. By knowing the thermal environment, the temperature of the outside surface of the spacecraft can be determined. This information, along with the propellant tank size and shape, and several other factors, allows the calculation of an expected boiloff rate.

For satellites or other spacecraft in Earth orbit, the thermal environment consists of three external sources of heat– energy from the Sun (solar flux), Earth-reflected heating (albedo times the incident solar flux), and Earth-emitted radiation, also called Earth infrared radiation, or simply Earth-IR. These are significant and can be calculated. At geostationary orbit (GEO), the values for Earth reflected heating and earth-IR drop off noticeably. At Earth-Moon L1, the values for Earth reflected heating and Earth-IR are almost non-existent. Some thermal analysts ignore the effects of Earth reflected heating and Earth-IR at GEO and L1 (S. Sutherlin, personal communication, April 22, 2015) but they are included here to be consistent.

In his book *Thermal Structures for Aerospace Applications*, Earl A. Thornton provides the method to calculate the thermal environment (Thornton, 1996, p.29-31). He notes that environmental heating rates depend on altitude and orientation of the spacecraft (See Figure 12). Assuming the Earth and Moon are coplanar with the Sun will simplify the calculations.

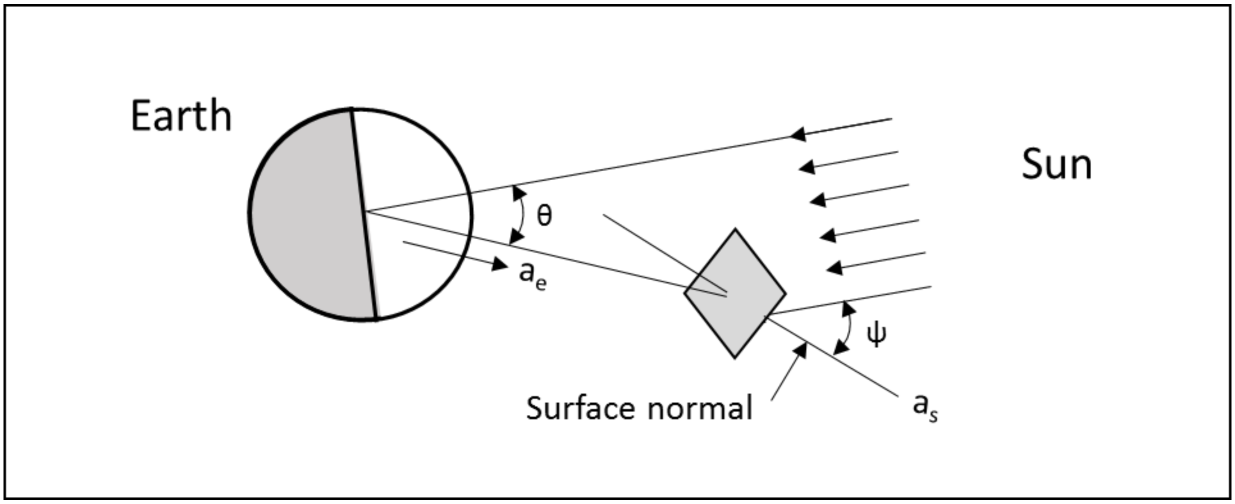

Figure 12. Orbital Heating Sources and Geometry

Tailoring the Equations

Figure 12 shows both the surface absorptivity for solar radiation, a_s, and Earth-emitted radiation, a_e. The solar constant is 1,367 W/m^2 at 1 astronomical unit (AU). The solar heat received by the spacecraft surface is given by

$$q_s = 1{,}367\ a_s \cos \psi \tag{7-1}$$

where ψ is the angle between the solar flux vector and the surface normal. (Thornton, 1996, p.29) At this point, we depart from Thornton slightly. Since our calculation for spacecraft surface temperature in Chapter VIII will account for the spacecraft surface absorptivity, we choose to set $a_s = 1$, so that it does not diminish the value of q_s. Secondly, since we assume the Earth and the Moon to be coplanar with the Sun, the value for $\psi = 0$ degrees, and the cosine of $\psi = 1$. Thus, the spacecraft is considered to be normal to the Sun for our calculations, and the value of the solar flux is unchanged at 1,367 W/m^2.

Thornton notes that the radiation emitted by the Earth (Earth-infrared) can be approximated by assuming the Earth to be a blackbody radiating at T_e = 289 K, and the radiation absorbed by the [spacecraft] surface can be expressed as:

$$q_e = \sigma\, T_e^4\, a_e\, F \tag{7-2}$$

where σ is Boltzmann's Constant 5.67051 x 10^{-8} W m^{-2} K^{-4}

a_e is the surface absorptivity for Earth-infrared radiation, and

F is the view factor.

The view factor (also called the shape factor or configuration factor) describes the fraction of the radiant energy that arrives at the surface (Thornton, 1996, p.30). The view factor is given by

$$F = \cos\lambda / H^2, \tag{7-3}$$

where λ = the angle between the surface normal and the heat flux

H = r/R, where R is radius of the Earth, and r is the distance from the center of the Earth to the spacecraft (Figure 13).

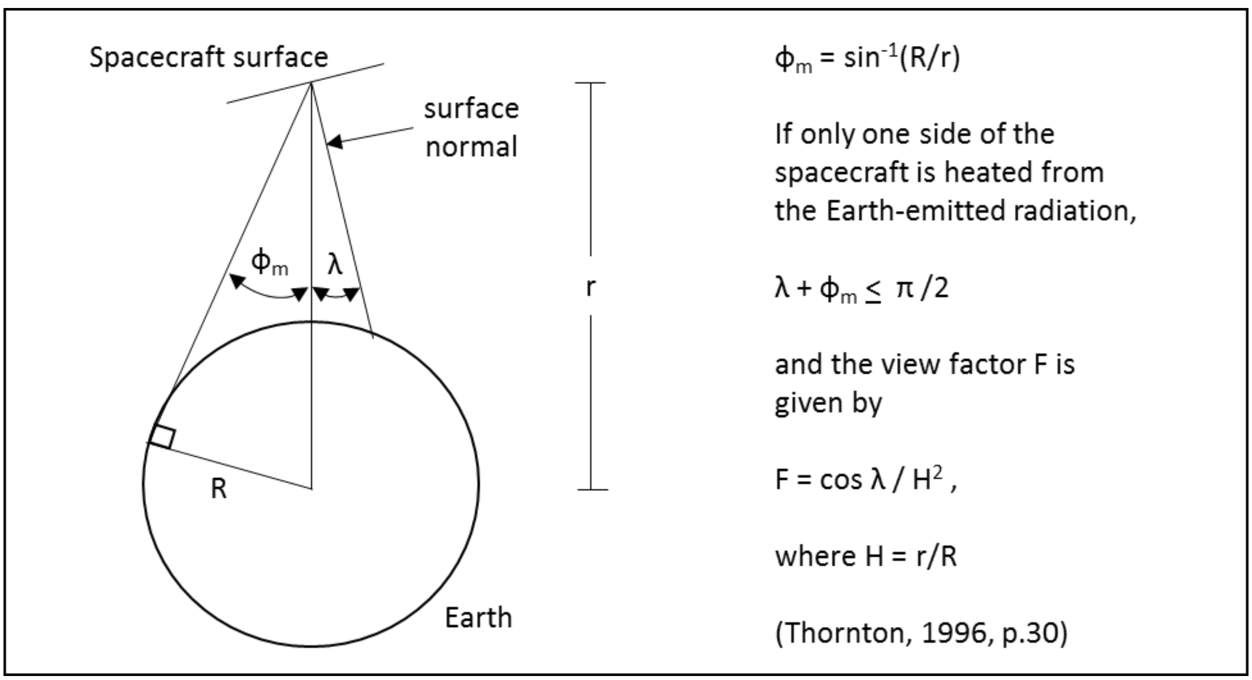

Figure 13. Earth-heating Geometry for View Factor Calculations

For this calculation, we again fall back to our thesis method and assumptions. Since we intend to calculate spacecraft surface temperature in conjunction with boiloff calculations and account for surface absorptivity there, surface absorptivity for this calculation is set to 1 so the value for q_e is not diminished. The spacecraft surface normal points to the center of the Earth; the angle λ is zero degrees, and the cosine is 1. Thus, the view factor F becomes

$$F = 1/H^2 \tag{7-4}$$

Thornton states that Earth reflected heating depends on the albedo factor (AF), and is defined as the fraction of the solar radiation striking the Earth that is reflected back into space. Earth reflected heating is described by:

$$q_a = 1{,}367\ AF\ a_s\ F \cos\theta \tag{7-5}$$

where θ is the reflection angle from the Earth to the spacecraft (shown in Figure 12). For this calculation, we chose an average Earth albedo of 0.367 provided by NASA/Jet Propulsion Laboratory (Planets and Pluto, 2008). Again, surface absorptivity is set at 1, and the angle θ = zero, and so the expression reduces to $q_a = 1{,}367\ (0.367)\ (F)$. Thus, our expressions for solar flux, Earth infrared radiation, and Earth reflected heating are:

Solar flux = 1,367 Watts/meter^2 (for all locations)

Earth infrared = σT_e^4 F, where T = 289 K, and $F = 1/H^2 = (R/r)^2$, and

Earth reflected heating = 1,367 (0.367) (F), where $F = 1/H^2 = (R/r)^2$

Subsequent Calculations – Earth Infrared

Earth infrared$_{LEO}$ = σT_e^4 F, where T = 289 K, and $F = (R/r)^2$

= $(5.67051 \text{ x } 10\text{-}8)\ (289)^4\ (6{,}378/6{,}778)^2$

= 350.3 W/m^2

$$\text{Earth infrared}_{GEO} = \sigma T_e^4 F\text{, where } T = 289 \text{ K, and } F = (R/r)^2$$

$$= (5.67051 \times 10\text{-}8)\ (289)^4\ (6{,}378/42{,}164)^2$$

$$= 9.1 \text{ W/m}^2$$

$$\text{Earth infrared}_{L1} = \sigma T_e^4 F\text{, where } T = 289 \text{ K, and } F = (R/r)^2$$

$$= (5.67051 \times 10\text{-}8)\ (289)^4\ (6{,}378/322{,}127)^2$$

$$= 0.16 \text{ W/m}^2$$

Subsequent Calculations – Earth Reflected Heating

$$\text{Earth Reflected Heating}_{LEO} = 1{,}367\ (0.367)\ (F)\text{, where } F = (R/r)^2$$

$$= (1367)\ (0.367)\ (6{,}378/6{,}778)^2$$

$$= 444.2 \text{ W/m}^2$$

$$\text{Earth Reflected Heating}_{GEO} = 1{,}367\ (0.367)\ (F)\text{, where } F = (R/r)^2$$

$$= (1367)\ (0.367)\ (6{,}378/42{,}164)^2$$

$$= 11.5 \text{ W/m}^2$$

$$\text{Earth Reflected Heating}_{L1} = 1{,}367\ (0.367)\ (F)\text{, where } F = (R/r)^2$$

$$= (1367)\ (0.367)\ (6{,}378/322{,}127)^2$$

$$= 0.20 \text{ W/m}^2$$

The values calculated for all sources are summarized in Table 14 below.

Table 14. Thermal Environment at LEO, GEO, and L1

Heat (Watts/m^2)	LEO	GEO	L1
- Solar constant	1,367	1,367	1,367
- Earth emitted infrared	350.3	9.1	0.16
- Earth reflected heating	444.2	11.5	0.20
Total (Watts/m^2) [1]	2,161.5	1,387.6	1,367.36

Notes:
[1] This represents the energy deposited on the cross section of the spacecraft propellant tanks.

CHAPTER VIII

CALCULATING BOILOFF LOSSES

Having established the mass of propellant required for the design reference missions, the implications for depot sizing and depot throughput, and the thermal environment in which these spacecraft will operate, we must now calculate anticipated propellant losses due to boiloff. First we will calculate the sizes of our spacecraft propellant tanks, since the configuration and surface area of the tanks influences boiloff. Then we will use the thermal environment data from the previous chapter to calculate the outside temperature of the spacecraft. Then we will use the "Modified Lockheed Model" to calculate boiloff rates. And lastly, we will use the time-of-flight values developed in Chapter V to figure anticipated losses for each of the segments in the architecture network diagram.

Oxidizer-to-Fuel Ratio for LH2/LO2 Rocket Engines

Stoichiometry would dictate the oxidizer-to-fuel ratio (O/F ratio) for a liquid hydrogen/liquid oxygen rocket engine should be 8:1 and the product of the combustion would be all water: $2H_2 + O_2 \rightarrow 2H_2O$

However, the O/F ratio used with these engines often does not include enough oxidizer for complete combustion. The primary reason that 6:1 is used (actually between 5 and 6) is that a considerable portion of the exhaust gas will be unburned hydrogen. Since the hydrogen molecules are lighter than water molecules, the exhaust velocity is greater, producing an increased specific impulse, I_{sp}. Huzel and Huang (1992) write:

Stoichiometric mixture ratio depends on the type of propellant used. Theoretical temperature and heat release are maximum at this ratio. In rocket engines, however, where the highest possible exhaust velocity is desired, optimum conditions often prevail at other than stoichiometric ratios…The lower the molecular weight, the higher the exhaust velocity, other things being equal. Analytical and experimental investigations will determine the optimum balance between energy release (heat) and composition (molecular weight) of the gas, a portion of which will consist of gasified but unburnt propellants. The optimum point may also be affected by the following:

- Stay time of the burning gas in the combustion chamber. Stay time is a function of combustion chamber volume and of gas volumetric flow rate. Complete combustion, even though desirable, requires a finite time, which will not be available unless the chamber is relatively large, and correspondingly heavy. (further text omitted)
- Cooling considerations. The temperatures resulting from stoichiometric or near-stoichiometric mixture ratios, dependent on propellant type, may impose severe demands on the chamber wall cooling system. A lower temperature may therefore be desired, and can be obtained by selecting a suitable ratio.
- Propellant density. Propellant density can make it profitable to deviate from the mixture ratio that yields optimum specific impulse. For example in the case of the LOX/LH2 propellant combination, where the density of the oxidizer is 16 times that of the fuel, vehicle manufacturers prefer to sacrifice some engine performance to obtain smaller tanks, and thus lower overall system weight.

> Typically, an engine mixture ratio of 6 is used for LOX/LH2. (Huzel & Huang, 1992, p.26)

They go on to write the vehicle [rocket] will be sized and tanked to conform to the chosen mixture ratio. (Huzel & Huang, 1992) For example, the shuttle external tank carried 629,340 kg of LO2 and 106,261 kg of LH2. Simple division yields an O/F ratio of 5.92:1.

Calculating Propellant Tank Sizes

Propellant tank sizes were based on the Δv values for a given mission. The rocket equation was used to estimate the fuel needed for the mission. That quantity of fuel was divided by 7 to establish the desired 6:1 oxidizer-to-fuel (O/F) ratio. Six sevenths of the mass was allocated to LO2, while one seventh was allocated to LH2. The masses for LO2 and LH2 were then divided by their respective densities to determine the volume of the bulk LO2 tank and the bulk LH2 tank. The formula for the volume of a spherical tank was then used to solve for the radius of each tank. Spherical tanks were used because the shape of the tanks is the same regardless of the volume. Using spherical tanks thus eliminated the shape and configuration of the fuel tanks as a factor in the subsequent boiloff calculations. Where canister tanks were to be employed, the canister size was essentially the size of the LO2 tank; three such tanks for LH2 were allocated for each LO2 tank.

<u>Example</u>: Calculate the tank sizes for a mission requiring 140,000 kg of propellant.

Applying a 6:1 O/F ratio, 140,000 kg of propellant breaks down to 120,000 kg of LO2 and 20,000 kg of LH2.

The volume of the LO2 tank is calculated by dividing the mass of the LO2 by the density of the LO2: 120,000 kg / 1,191.6 kg/m^3 = 100.7 m^3

The volume of the LH2 tank is calculated by dividing the mass of the LH2 by the density of the LH2: 20,000 kg / 70.99 kg/m^3 = 281.7 m^3

Substituting into the formula for the volume of a sphere yields the radius of each tank:

Volume of a sphere = $4/3\ \pi\ r^3$, where r = the radius of the sphere

Solving for r, $r = (3/4 \text{ x volume}/ \pi)^{1/3}$

For the LO2 tank, $r = (3/4 \text{ x } 100.7/ \pi)^{1/3}$ = 2.89 meters

For the LH2 tank, $r = (3/4 \text{ x } 281.7/ \pi)^{1/3}$ = 4.07 meters

For sizing the canister tanks, a different method was used. The volume of the tank was calculated based on the knowledge that three LH2 tanks would be used for each LO2 tank. Thus,

$(x\ m^3)(1191.6\ kg/m^3) + (3x\ m^3)(70.99\ kg/m^3) = 140{,}000$ kg

1191.6 x + 212.97 x = 140,000

x = 140,000/1404.57 = 99.67 m^3

Again using the volume of a sphere, $r = (3/4 \text{ x } 99.67/ \pi)^{1/3}$ = 2.88 meters

For the MCV bulk fuel tanks, the dimensions were calculated based on the 6:1 O/F ratio as before, and the formula for the volume of a cylinder, with the knowledge the vehicle diameter is 10 meters and the maximum fuel capacity is 250,000 kg (Kyle, 2010). Applying a 6:1 O/F ratio, 250,000 kg of propellant breaks down to 214,286 kg of LO2 and 35,714 kg of LH2.

The volume of the LO2 tank is calculated by dividing the mass of the LO2 by the density of the LO2: 214,286 kg / 1,191.6 kg/m^3 = 179.8 m^3

The volume of the LH2 tank is calculated by dividing the mass of the LH2 by the density of the LH2: 35,714 kg / 70.99 kg/m^3 = 503.1 m^3

The volume of a cylinder = (πr^2)(h), where r is the radius of the tank and h is he height. Solving for h, h = volume/(πr^2)

The height of the LO2 tank is 179.8/25 π = 2.29 meters.

The height of the LH2 tank is 503.1/25 π = 6.41 meters.

Calculated tank sizes for all vehicles is shown in Table 15 below. All propellant tanks are spherical except for the cylindrical bulk tanks on the MCV.

Table 15. Calculated Propellant Tank Sizes

<table>
<tr><th>Delivery Location</th><th>Delivery Method</th><th>LTV Propellant</th><th>LTV Payload</th><th>CSSV</th><th>MCV</th></tr>
<tr><td>LEO</td><td>(ALL)</td><td colspan="4">The LTV cannot service the CSSV, MCV, or depot in LEO. The round trip from the Moon to LEO takes more fuel than it carries.</td></tr>
<tr><td rowspan="2">GEO</td><td>BF</td><td>LH2: 4.80 m
LO2: 3.41 m</td><td>LH2: 1.91 m
LO2: 1.36 m</td><td>LH2: 3.75 m
LO2: 2.66 m</td><td>LH2: 10 x 6.36m
LO2: 10 x 2.29m</td></tr>
<tr><td>CX</td><td colspan="4">LH2/LO2: 1.35 m</td></tr>
<tr><td rowspan="2">L1</td><td>BF</td><td>LH2: 3.92 m
LO2: 2.79 m</td><td>LH2: 3.85 m
LO2: 2.73 m</td><td>LH2: 3.34 m
LO2: 2.37 m</td><td>LH2: 10 x 6.41m
LO2: 10 x 2.29m</td></tr>
<tr><td>CX</td><td colspan="4">LH2/LO2: 1.35 m</td></tr>
</table>

Calculating Spacecraft Surface Temperature

The calculated values for the solar constant, Earth reflected heating, and earth infrared for each of the three depot locations from Chapter VII become part of the calculation to determine the surface temperature of the spacecraft. We start with a basic equation provided by Wertz and Larson (Wertz and Larson, 1999, p.435):

$$\sigma T^4 = (\alpha/\varepsilon)(S) \times (A_p/A) \tag{8-1}$$

Here, Wertz and Larson focus on the solar flux, but do not address Earth reflected heating and Earth infrared. To address these sources, we adapt the equation by adding terms for reflected heating and Earth infrared in equation 8-2 and simplify the equation to yield equation 8-3. (S. Sutherlin, personal communication, April 22, 2015) Notice the value for Earth reflected heating is modified by α/ε in the same manner as the solar flux term S, but the value for Earth infrared is not. The ratio of absorptivity to emissivity is used to determine that portion of the broad spectrum energy which contributes to heating the outside of the spacecraft. For the infrared band, emissivity is the multiplying factor for both emission and absorption, and the term simplifies to E. The A_p/A term defines the portion of the propellant tank being heated.

$$\sigma T^4 = [(1/\varepsilon)(\alpha(S+RH) + \varepsilon(E))] \times (A_p/A) \tag{8-2}$$

$$\sigma T^4 = [(\alpha/\varepsilon)S + (\alpha/\varepsilon)RH + (\varepsilon/\varepsilon)E] \times (A_p/A)$$

$$\sigma T^4 = [(\alpha/\varepsilon)(S) + (\alpha/\varepsilon)(RH) + E] \times (A_p/A) \tag{8-3}$$

where T = outside temperature of the spacecraft (K)

σ = Boltzmann's constant = 5.67051×10^{-8} W/m^2 K^{-4}

α = absorptivity *(= 0.14 for outer layer of MLI)*

ε = emissivity *(= 0.60 for outer layer of MLI)*

S = solar flux (1,367 W/m^2)

RH = Earth reflected heating

E = Earth infrared

A_p = projected area of the propellant tank

A = total surface area of the propellant tank

Solving the equation for T gives the surface temperature of the spacecraft in Kelvin, which is to say the surface temperature of the propellant tanks. We assume the propellant tanks for the various spacecraft (CSSV, MCV, and LTV) are exposed to space, except for layers of multi-layer insulation (MLI). The propellant tanks are not covered by any sort of enclosure.

It is important to mention the absorptivity and emissivity values for the outer layer of multi-layer insulation (MLI). The outer layer of MLI is chosen such that it has a low absorptivity but a relatively high emissivity. Thus, the outer layer of MLI reflects as much of the incoming energy as possible, but the high value for emissivity means the MLI allows as much heat as possible to escape. The values for absorptivity and emissivity shown are based on the recommendations of the Advanced Concepts Office at NASA/Marshall Space Flight Center (S. Sutherlin, personal communication, April 22, 2015) and correspond to the specifications of Sheldahl Aluminum-coated (one side) Fluoro ethylene propylene (FEP) (Sheldahl, 2015, p.53).

The term A_p/A represents that portion of the propellant tank which is receiving the energy. That the energy comes from opposite directions is not a concern, since the propellant tanks are essentially homogenous and only contain the cryogenic propellant. A is the surface area of the tank, while A_p is the projected area of the tank. Since we assume the Earth, spacecraft, and Sun to be coplanar, the projected area of a spherical tank is the area of a circle having the same radius as the tank, and the projected area of a cylindrical tank is the area of a rectangle having a width equal to the diameter of the propellant tank and a length equal to the height of the tank.

The values calculated for the surface temperature of the propellant tanks for the DRM vehicles are given in Table 16. The differing values for the LH2 and LO2 tanks of the MCV stem from the difference in length of the two tanks. The value of A_p/A for the (longer) LH2 tank is 0.1781, while the value for the LO2 tank is 0.1. A_p/A for all spherical tanks is 0.25.

Table 16. Calculated Surface Temperatures for DRM Propellant Tanks

Location	Surface Temperature for All Spherical Tanks for CSSV, MCV, LTV (K)	Surface Temperature for Bulk Propellant Tanks for MCV (K)
LEO	241.608	LH2: 221.981 LO2: 192.129
GEO	195.413	LH2: 179.538 LO2: 155.395
L1	193.681	LH2: 177.946 LO2: 154.017

Modified Lockheed Model

It was mentioned in Chapter I that spray-on foam insulation (SOFI) has no insulating value in space. For this reason, blankets constructed of multilayer insulation (MLI) are used to insulate propellant tanks in space. The density of the individual layers (described as layers per centimeter, the “thickness” of the blanket) can be varied, often using more layers near the warm boundary (outside the tank) and fewer layers near the cold boundary – the surface of the tank itself. This gives rise to the term variable density multilayer insulation, or VD-MLI. Hastings, Hedayat, and Brown note there are two analytical models for predicting the performance of VD-MLI blankets. One is a layer-by-layer analysis which is likely cumbersome. The other model is an empirical model (equation 8-4) developed over the years known as the Modified Lockheed Model (Hastings, Hedayat, & Brown, 2004).

The Modified Lockheed Model considers three heat transfer mechanisms – solid conduction, radiation between blanket layers, called shields, and gas conduction. The model consists of four terms which are added together. The first two terms describe the solid conduction. The third term describes the radiation between blanket layers, and the fourth term describes the gas conduction. The output of the Modified Lockheed Model is q, the rate of heat transfer through the layers of insulation into the fuel tank in W/m^2.

$$q = 0.00024*(0.017+7E\text{-}6(800-T) +0.0228*\ln(T))*(N^*)^{2.63}(T_h-T_c)/N_s \quad (8\text{-}4)$$
$$+ 4.944E\text{-}10*\varepsilon*(T_h^{4.67}-T_c^{4.67})/N_s + 1.46E4*P*(T_h^{0.52}-T_c^{0.52})/N_s$$

where

q = heat transfer rate in W/m^2

ε = emissivity of the inner layers of MLI (here = 0.035)

T_h = temperature on outside tank surface (K)

T_c = propellant temperature (20 K for LH2, 80 K for LO2)

T = $(T_h+T_c)/2$

N^* = number of layers/cm of MLI

N_s = number of layers of MLI, and

P = pressure between the layers of MLI (Torr)

The surface temperature of the spacecraft calculated earlier is one input to the model. Other important inputs are N_s -- the number of layers of MLI, N* -- the number of layers per centimeter of MLI, and ε -- the emissivity of those layers.

Sixty layers of MLI were chosen for the thesis calculations. In their paper *Cryogenic Thermal System Analysis for [an] Orbital Propellant Depot*, authors Patrick Chai and Alan Wilhite demonstrated analytically that with 60 layers, the rate of cryogenic boiloff stabilizes to approximately 0.5 – 1.0% per month for LO2, and approximately 2.5 – 5.0% per month for LH2. (Chai and Wilhite, 2013) Furthermore, their results pointed to the density of the MLI blankets (number of layers per centimeter) as being important, with more space between the layers being better. For this effort, Steven Sutherlin suggested a density of 10 layers per centimeter, and using inner layers of MLI with a low emissivity (S. Sutherlin, personal communication, May 7, 2015).

The low-emissivity MLI limits the amount of infrared radiation transmitted from layer to layer. Aluminum-coated (two sides) polyethylene terephthalate (PET, commonly known as Mylar) (Sheldahl, 2015, p.19) with an emissivity of 0.035 was chosen for the thesis calculations.

Spreadsheet software was used to perform the calculations. The inputs and terms of the model were arrayed from left to right, while the locations of interest (LEO, GEO, and L1) were arrayed from top to bottom. Within each location, calculations were performed for each configuration of propellant tanks for both liquid hydrogen and liquid oxygen.

The output of the Modified Lockheed Model is q, the rate of heat transfer through the layers of insulation into the fuel tank in W/m^2. The total heat transfer (Watts) is calculated by multiplying the rate of heat transfer times the surface area of the tank. Then, dividing the total heat transfer by the heat of vaporization for the cryogenic fluid in the tank (in Joules/kilogram) yields the rate of boiloff in kilograms/second. (A.Wilhite, personal communication, March 25, 2014). The desired boiloff rate in kilograms/hour is obtained by multiplying the kg/sec rate x 3,600 seconds/hour. Complete calculations are provided in the compact disk. Table 17 provides the range of boiloff rates across all tank configurations for each location.

Table 17. Range of Boiloff Rates Across All Tank Configurations

Cryogen	LEO (400 km)	GEO (42,164 km)	L1 (322,127 km)
LH2 boiloff (kg/hr)	0.0164 - 0.2044	0.0097 – 0.1268	0.0095 – 0.1245
LO2 boiloff (kg/hr)	0.0296 – 0.1437	0.0157 – 0.1002	0.0153 – 0.0977

1- Rate calculated for across all tank configurations, using the modified Lockheed Model.
2- Included 60 layers of multi-layer insulation (MLI)
3- Solar constant, earth reflected heating, and earth infrared radiation included for each orbit.

Chilldown Losses

One motivation for considering the exchange of canister propellant tanks rather than transferring bulk cryogens in microgravity is that transferring bulk cryogens in microgravity has not yet been demonstrated (Chato, 2005). Exchanging canister tanks would be a way to bypass that task. Another reason to consider canister exchange is due to “chilldown loss” (P. McRight, personal communication, April 17, 2015). In preparation for the fluid transfer, it is necessary to chill the pipe through which the cryogen will move, or else the heat of the pipe will cause the cryogen to boil. The typical procedure is to partially fill the transfer pipe with the cryogen. It will boil, of course, but in doing so cools the pipe. The procedure is then repeated to complete cooling the pipe. Then the planned fluid transfer can be initiated. The two partial releases into the transfer pipe equate to having filled the pipe completely one time. So, the chilldown loss is considered to be the volume of the pipe times the density of the cryogen. This loss would be incurred for every transfer. For example, transferring propellant from the LTV to a depot would incur a chilldown loss, and transferring propellant from the depot to a customer vehicle would incur an additional chilldown loss.

For the thesis effort, it was necessary to choose the dimensions for a transfer pipe. The pipe used with the Space Shuttle External Tank (ET) was 17 inches (0.43 meters) in diameter. Comparing the volumes of the ET to the volumes of the propellant tanks for the design reference missions, a diameter of 0.1 meters was chosen as being proportional.

Regarding the length, the Shuttle Remote Manipulator System (also known as the Canadarm) was about 15 meters long. It is conceivable that a vehicle attempting to dock with a depot (to either deliver or obtain fuel) would be grasped by similar arms. These arms would then telescope or retract and pull the visiting vehicle toward to the depot, until such time as the mating of the transfer pipes between the vehicle and depot was accomplished. Envisioning this scenario, the transfer pipe would need to be shorter than the grappling arms, so a length of 10 meters was chosen.

The volume of the notional transfer pipe is then:

$$\begin{aligned} \text{Volume} &= \pi r^2 h \\ &= \pi (0.05 \text{ meters})^2 \times 10 \text{ meters} \\ &= 0.0785 \text{ m}^3 \end{aligned}$$

For liquid hydrogen, the chilldown loss would be

$$\text{Chilldown loss}_{LH2} = 0.0785 \text{ m}^3 \times 70.99 \text{ kg/m}^3 = 5.57 \text{ kg per transfer.}$$

For liquid oxygen, the chilldown loss would be

$$\text{Chilldown loss}_{LO2} = 0.0785 \text{ m}^3 \times 1911.6 \text{ kg/m}^3 = 93.6 \text{ kg per transfer.}$$

Boiloff and Chilldown Losses for Each Network Segment

Once the boiloff rates for the different tank sizes were calculated for the different locations, the next step was to calculate boiloff and chilldown losses due to boiloff for each candidate architecture using the time-of-flight values developed in Chapter V. For example, if a conjunction class trajectory from Earth to Mars takes 288 days, that value (converted to hours) would be multiplied by the appropriate boiloff rates to determine the mass of LH2/LO2 lost during the journey. Calculations performed also included boiloff during any idle time. For example, the CSSV services 10 satellites during each mission, but loiters at the ISS between missions. Boiloff incurred during this idle time between missions was also captured.

One situation that arises has to do with calculating boiloff when a vehicle is moving from one orbit to another, such as the LTV delivering fuel to geostationary orbit. In such cases, the boiloff rates were used for that region in which the preponderance of the maneuver time, i.e., time of flight, was spent. Thus, for the LTV delivering fuel to GEO, the boiloff rates for L1 were applied.

For the CSSV refueling at L1, boiloff rates for all three orbits were used. The LEO rates was used when the CSSV was idle at the ISS. The GEO rates was used when servicing satellites. And lastly, the L1 rates was used when maneuvering to L1, i.e., beyond GEO.

For the MCV, the LEO rates was used in LEO, the GEO rates was used if maneuvering to GEO to refuel, and the L1 rates used for the transit to Mars.

Separate spreadsheets were constructed for each orbiting depot location – GEO and L1. (Recall that a depot located in LEO is unworkable, because the mass of fuel required for the LTV to travel from the Moon to LEO and back greatly exceeds the lift capacity of the vehicle.) Each spreadsheet has different rows allocated to the various network segments. For the LTV to deliver propellants to L1, only two rows are needed – one for LH2 and one for LO2, and only the boiloff rates for L1 are used. For MCV refueling in GEO, six rows are used – two for the time spent in LEO, two rows for the maneuvering to GEO to refuel, and two rows for the transit to Mars. In each case the numbers of propellant tanks are accounted for, whether bulk fuel tanks or canister tanks are used. Where bulk propellants are used, the chill-down losses are shown. Finally, the number of trips the LTV makes to supply a depot in GEO is much greater than the number of trips to supply a depot at L1. The spreadsheets account for the number of trips in calculating the total anticipated losses.

Figures for propellant consumption, propellant loss due to boiloff, and chilldown losses were entered manually into a separate spreadsheet. Separate rows were used for each architecture, and a separate column for each term in the objective function. After completing the table, the rows (architectures) were sorted from smallest to largest in terms of total propellant.

CHAPTER IX

RESULTS

Calculated Propellant Consumption and Losses

The candidate architectures and their respective propellant consumption and losses are presented in Table 18 below. The architectures are sorted so the most efficient are at the top and the least efficient are at the bottom. The results for each architecture are discussed below in order from most efficient to least efficient, along with the total consumption and losses for each.

Architecture 1-2-8

Architecture 1-2-8 provides for the bulk shipment of water to a depot at L1. By virtue of shipping bulk fluid, it can take advantage of the LTV's maximum payload mass. There is no boiloff or chilldown losses for the water as the LTV payload. However, both the CSSV and the MCV will incur chilldown losses when transferring bulk propellant, as well as boiloff losses during their respective missions. Total propellant consumption/losses = 1,228,254 kg.

Architecture 1-4-14

Architecture 1-4-14 provides for the bulk shipment of propellants to a depot in L1. Consumption and losses for this candidate are the same as for Architecture 1-2-8, except that the payload propellant is subject to boiloff losses in transit to the depot, and chilldown losses to transfer from the LTV to the depot. CSSV and MCV losses are unchanged. Total propellant consumption/losses for this candidate = 1,228,905 kg.

Table 18. Propellant Consumption and Loss by Candidate Architecture

Candidate Architecture	LTV Fuel Consumed (P_{LTV})	LTV Boiloff Losses (B_{LTV})	LTV P/L Boiloff ($B_{P/L}$)	LTV Chilldown Losses ($C_{P/L}$)	CSSV Fuel Consumed (P_{CSSV})	CSSV Chilldown Losses (C_{CSSV})	CSSV Boiloff Losses (B_{CSSV})	MCV Fuel Consumed (P_{MCV})	MCV Chilldown Losses (C_{MCV})	MCV Boiloff Losses (B_{MCV})	Architecture Total Propellant (kg)
1-2-8 BF water L1	631,600	96	0	0	466,818	596	616	126,978	99	1,450	1,228,254
1-4-14 BF, fuel, L1	631,600	96	55	596	466,818	596	616	126,978	99	1,450	1,228,905
1-5-17 CX, fuel, L1	673,551	115	118	0	466,818	0	1,528	126,978	0	3,096	1,272,203
1-3-11 CX water, L1	676,554	115	0	0	466,818	0	1,528	126,978	0	3,096	1,275,088
1-4-20 BF, Direct Delivery,L1	715,005	96	74	0	466,818	595	616	126,978	198	1,450	1,311,830
1-5-23 CX, Direct Delivery,L1	769,129	134	150	0	466,818	0	1,528	126,978	0	3,096	1,367,832

Table 18. cont.

Candidate Architecture	LTV Fuel Consumed (P_{LTV})	LTV Boiloff Losses (B_{LTV})	LTV P/L Boiloff ($B_{P/L}$)	LTV Chilldown Losses ($C_{P/L}$)	CSSV Fuel Consumed (P_{CSSV})	CSSV Chilldown Losses (C_{CSSV})	CSSV Boiloff Losses (B_{CSSV})	MCV Fuel Consumed (P_{MCV})	MCV Chilldown Losses (C_{MCV})	MCV Boiloff Losses (B_{MCV})	Architecture Total Propellant (kg)
1-2-7 BF water GEO	12,246,445	3,288	0	0	661,374	596	743	102,740	99	1,432	13,016,719
1-5-16 CX, fuel, GEO	12,246,445	3,288	352	0	661,374	0	1,937	102,740	0	2,445	13,018,582
1-4-13 BF, fuel, GEO	12,246,445	3,288	278	5,258	661,374	596	738	102,740	99	1,432	13,022,248
1-3-10 CX water GEO	16,227,386	4,591	0	0	661,374	0	1,937	102,740	0	2,445	17,000,474

Architecture 1-5-17

This architecture provides for the delivery of propellants in canisters to a depot at L1. The situation for water in Architecture 1-3-11 is replicated for propellants in architecture 1-5-17. Adhering to an O/F ratio of 6, the LTV can carry 8 LO2 canisters and 24 LH2 canisters, totaling 116,186 kg, less than the LTV max payload. The fuel required is 124,152 kg; for six trips is 744,912 kg. No chilldown losses are incurred, but boiloff is increased over bulk fluid shipments, due to the increased cumulative surface area of the canisters. Total propellant consumption/ losses for this candidate is 1,272,203 kilograms.

Architecture 1-3-11

This architecture provides for the delivery of water in canisters to a depot at L1. Recall from Table 15 that a canister of radius 1.35 meters is used for all DRM vehicles to all locations. The volume of these canisters is 10.34 m^3. For delivery to L1, the maximum payload for the LTV is 119,275 kilograms. The density of water is 1,000 kg/m^3. Therefore, the mass of each canister is 10,340 kilograms. The LTV can only carry 11 canisters for a total payload of 113,740 kilograms; thus six LTV flights are required. Since the payload is less than the LTV maximum, the fuel required for this payload is 122,435 kilograms; for six trips is 734,610 kilograms. Boiloff of the LTV's own propellant is 96 kilograms. There are no boiloff or chilldown losses for the water. The propellant mass for the CSSV is for 6 missions. There is no chilldown loss, but boiloff losses are larger due to the greater surface area of the canister propellant tanks. These losses carry over to the MCV as well. Total propellant consumption/losses for this candidate is 1,275,088 kilograms.

Architecture 1-4-20

Architecture 1-4-20 provides for the direct delivery of bulk propellants from the Moon to the CSSV and MCV. This generally provides an advantage in terms of efficiency, because the LTV delivers only the fuel that is required (generally less than its maximum payload), and uses less propellant to deliver it. However, in this case, the LTV must make 7 flights to service the CSSV and MCV directly, less efficient than Architecture 1-4-14. In addition, the CSSV and MCV must split the payload of the last LTV flight, and thus incur an additional fuel transfer – and chilldown loss (eight chilldowns for seven flights). Total propellant consumption/losses for this candidate = 1,311,830 kg.

Architecture 1-5-23

This architecture provides for the direct delivery of propellant canisters from the Moon to the CSSV and MCV. This architecture requires seven flights to deliver the total number of canisters needed by the CSSV and MCV, with the last flight splitting its payload between the two DRM vehicles. (This makes little practical sense, but it can be done.) Total propellant consumption/losses for this candidate is 1,367,832 kilograms.

Architecture 1-2-7

This architecture provides for the bulk shipment of water from the Moon to a depot in GEO. This is the first of the architectures involving a depot located in GEO. The payload for the LTV is greatly reduced, so many more flights are needed to support the depot – 53 – and propellant expenditures to support the depot increase by an order of magnitude. No payload boiloff or chilldown losses are incurred for the LTV. Total propellant consumption/losses for this candidate is 13,016,719 kilograms.

Architecture 1-5-16

This architecture provides for the bulk shipment of propellant in canisters to a depot in GEO. Boiloff losses are incurred for the propellant during shipment to the depot, and after transfer to the DRM vehicles. Since the method of transfer is canister exchange, there are not losses due to chilldown. Total consumption/losses for this candidate is 13,018,582 kilograms.

Architecture 1-4-13

This architecture provides for the bulk shipment of bulk propellants to a depot in GEO. Propellant boiloff is less than for canisters (reduced overall surface area), but propellant losses are overtaken by chilldown losses for the number of flights needed to support the depot. Total propellant consumption/losses for this candidate is 13,022,248 kilograms.

Architecture 1-3-10

This architecture provides for the bulk shipment of water in canisters from the Moon to a depot in GEO. This is by far the least efficient of the candidate architectures. The volume of each canister is 10.34 m^3. The density of water is 1,000 kg/m^3. So each canister of water is 10,340 kg, far less than the 14,520 kg payload of the LTV to GEO. A grand total of 74 flights are then needed to support the depot. So the mass of propellant necessary for this architecture for the operation of the LTV(s) greatly increases. Total propellant consumption/losses for this candidate is 17,000,474 kilograms.

Architecture Statistics

With any sort of project that involves the expenditure of resources, it is useful to compile some statistics that will give insight into the operation. The following statistics were compiled and are listed in Table 19 below: LTV losses as a percentage of propellant used, CSSV losses as a percentage of propellant used, MCV losses as a percentage of propellant used, boiloff as a percentage of total fuel consumed, and boiloff as a percentage of total fuel shipped.

Table 19. Propellant Loss Statistics

Candidate Architecture	LTV losses % of fuel consumed	CSSV losses % of fuel consumed	MCV losses % of fuel consumed	Boiloff % of fuel consumed	Boiloff % of fuel shipped
1-2-8 BF H2O L1	0.015%	0.260%	1.220%	0.233%	0.481%
1-4-14 BF prop L1	0.015%	0.260%	1.220%	0.286%	0.591%
1-5-17 CX prop L1	0.015%	0.327%	2.438%	0.364%	0.821%
1-3-11 CX H2O L1	0.016%	0.327%	2.438%	0.357%	0.798%
1-4-20 BF DD L1	0.013%	0.260%	1.298%	0.231%	0.510%
1-5-23 CX DD L1	0.017%	0.327%	2.438%	0.360%	0.826%
1-2-7 BF H2O GEO	0.027%	0.203%	1.491%	0.047%	0.806%
1-5-16 CX prop GEO	0.027%	0.293%	2.380%	0.062%	1.050%
1-4-13 BF prop GEO	0.027%	0.202%	1.491%	0.090%	1.530%
1-3-10 CX H2O GEO	0.028%	0.293%	2.380%	0.053%	1.174%

LTV losses as a percentage of propellant used

LTV losses as a percentage of propellant used ranged from 0.013% - 0.028%. The smallest losses as a percentage of propellant consumed is for LTV flights to L1, where both the least fuel was used and the time of flight was the shortest. The largest percentage of losses as a percentage of propellant consumed was for deliveries to GEO, where the increased number of flights and increased time of flight led to greater losses.

CSSV losses as a percentage of propellant used

CSSV losses as a percentage of propellant used ranged from 0.202% - 0.327%. The smallest percentage of losses as a percentage of propellant consumed was for the depot location in GEO, where the largest amount of propellant was consumed (fraction is smaller because denominator is larger). The largest percentage of losses as a percentage of propellant consumed was for depot locations at L1.

MCV Losses as a percentage of propellant used

MCV losses as a percentage of propellant used ranged from 1.491% - 2.438%. The smaller percentages were for architectures using bulk fuel transfer, while the larger percentages were for architectures using canister exchange. The increased surface area of the canisters – leading to increased boiloff – exceeded losses due to chilldown for the bulk transfer.

Boiloff as a percentage of total fuel consumed

Boiloff as a percentage of total fuel consumed ranged from 0.047% - 0.286%. The smallest percentage of losses as a percentage of propellant consumed was for the depot location in GEO, where the largest amount of propellant was consumed (fraction is smaller because denominator is larger). The largest percentage of losses as a percentage of propellant consumed was for depot locations at L1.

Boiloff as a percentage of total fuel shipped

Boiloff as a percentage of total fuel shipped ranged from 0.481% - 1.530%. The smallest percentage occurred for shipments to L1, where the fuel shipped was the least, and the fuel losses were smaller. The larger percentages were almost exclusively for shipments to GEO, where the mass of propellant shipped was greater, but the losses were greater as well.

CHAPTER X

SENSITIVITY ANALYSES

Having completed the main analyses, it can be instructive to perform sensitivity analyses to see how the results change. Two areas were selected for investigation – adding a second engine to the Lunar Tanker Vehicle (which we will note as LTV2), and investigating the results with fewer numbers of multilayer insulation (MLI). (Sixty (60) layers of insulation were used in the previous boiloff calculations.)

LTV with Two Engines (LTV2)

The results of the previous calculations indicated that 53 flights of the LTV were needed to support a depot in GEO. Each flight consumed a large mass of propellants, but was only able to deliver a fraction of that amount. Also, the LTV was able to service the CSSV at L1 with a single flight, but not the MCV. This lack of performance suggests the LTV was underpowered, and so an LTV with two engines was investigated.

To "create" the LTV2, it was assumed the vehicle structure would need to increase. The mass of the vehicle structure was increased from 20,000 kg to 30,000 kg. To that structure two J-2X engines were added, each with a mass of 2,470 kg, for a total dry mass of 34,940 kg. It was then necessary to calculate the maximum weight of the vehicle and the portion of the total mass that could be allocated to vehicle propellant or payload propellant. For the LTV2:

- Thrust for two J-2X engines is 2,614 kN (kiloNewtons)
- Surface gravitational acceleration for the Moon is 1.62 m/s^2

- Dry mass = 30,000 kg + 4,940 kg engines = 34,940 kg

As noted in Chapter IV, the optimum thrust-to-weight ratio for ascent/descent for lunar vehicle is about one-half Earth's gravitational acceleration, which equates to a TWR of 3. Setting the thrust-to-weight ratio at 3 yields:

$TWR = F_{thrust}/m \times g_{Moon}$

$3 = 2{,}614{,}000\ N / (m \times 1.62\ m/s^2)$

Solving for m:

$m = 2{,}614{,}000\ N / (3 \times 1.62)$

$= 536{,}142$ kg

Subtracting the vehicle dry mass yields the mass that can be allocated to vehicle propellant or propellant to be delivered to a customer:

536,142 – 34,940 = 501,202 kg

The many calculations that were performed for the main thesis effort were then repeated for LTV2. The first of these was to determine the fuel consumption and maximum payloads for LTV2 (Table 20). As with the LTV, the mass of fuel necessary for the LTV2 to fly from the

Table 20. Fuel Consumption and Maximum Payload for LTV2

Vehicle	Depot in LEO	Depot in GEO	Depot at L1
LTV2 Fuel Required (kg)	927,447 [1]	421,576	231,255
LTV2 Payload (kg)	---	44,686	235,100

Notes:
[1] LTV2 fuel required to deliver in LEO is greater than its total lift capacity.

Moon to low Earth orbit and return is greater than the lift capacity of the vehicle. The payload to GEO has improved, but the propellant expended to deliver to GEO far exceeds the payload. Only at L1 does the LTV2 deliver more fuel than it consumes.

But the increase in payload improves the capability to supply the depot (Table 21). While the LTV required 53 trips to service the depot in GEO, the LTV2 requires only 18. While the LTV required five trips to service the depot at L1, the LTV2 requires only three.

Table 21. LTV2 Flights to Supply the Depot

Depot Location	Mass Required per Six Months (kg)	Mass LTV can deliver per flight (kg)	LTV Flights needed to service the depot
LEO[1]	1,653,401	---	---
GEO	764,114	44,686	17.10 →18
L1	593,796	235,100	2.53 → 3
Notes: [1]LTV2 cannot support a depot located in LEO.			

Since the LTV2 is larger than the LTV, propellant tank sizes have also increased (Table 22).

Table 22. LTV2 Calculated Propellant Tank Sizes

Delivery Location	Delivery Method	LTV Propellant	LTV Payload	CSSV	MCV
LEO	(ALL)	The LTV cannot service the CSSV, MCV, or depot in LEO. The round trip from the Moon to LEO takes more fuel than it carries.			
GEO	BF	LH2: 5.86 m LO2: 4.16 m	LH2: 2.78 m LO2: 1.97 m	LH2: 3.75 m LO2: 2.66 m	LH2: 10 x 6.36m LO2: 10 x 2.29m
	CAN	LH2/LO2: 1.35 m			
L1	BF	LH2: 4.80 m LO2: 3.41 m	LH2: 4.83 m LO2: 3.43 m	LH2: 3.34 m LO2: 2.37 m	LH2: 10 x 6.41m LO2: 10 x 2.29m
	CAN	LH2/LO2: 1.35 m			

The ability of the LTV2 to service the design reference missions is given in Table 23. Like the LTV, it cannot service a depot or vehicles in LEO. The fuel required to fly to LEO and return to the Moon is greater than the vehicle's lift capacity. LTV2 can only service a depot in GEO, since its payload is too small to service the other vehicles. Only at L1 does the LTV2 deliver more fuel than it consumes, and only at L1 could the LTV2 service the CSSV and/or MCV directly.

Table 23. LTV2 Capacity to Service Design Reference Missions

Vehicle	Fuel Needed for mission (kg)	Mass LTV can deliver (kg)	Remarks
Commercial Satellite Servicing Vehicle (CSSV)			
Depot in LEO	243,621	---	LTV cannot service CSSV or depot in LEO.
Depot in GEO	110,229	44,686	LTV capacity is less than fuel required; impractical to service CSSV directly.
Depot in L1	77,803	235,100	LTV capacity is greater than fuel required; can service the depot, or the CSSV directly.
Mars Cargo Vehicle (MCV)			
Depot in LEO	191,075	---	LTV cannot service MCV or depot in LEO.
Depot in GEO	102,740	44,686	LTV capacity is much less than fuel required; impractical to service MCV directly.
Depot in L1	126,978	235,100	LTV capacity is greater than fuel required; can service MCV directly.

Propellant consumption and loss by candidate architecture is given in Table 24. It is noteworthy that total propellant consumption is much the same as for the LTV, while boiloff and chilldown values declined because fewer flights were required.

Table 24. LTV2 Propellant Consumption and Loss by Candidate Architecture

Candidate Architecture	LTV Fuel Consumed (P_{LTV})	LTV Boiloff Losses (B_{LTV})	LTV P/L Boiloff ($B_{P/L}$)	LTV Chilldown Losses ($C_{P/L}$)	CSSV Fuel Consumed (P_{CSSV})	CSSV Chilldown Losses (C_{CSSV})	CSSV Boiloff Losses (B_{CSSV})	MCV Fuel Consumed (P_{MCV})	MCV Chilldown Losses (C_{MCV})	MCV Boiloff Losses (B_{MCV})	Architecture Total Propellant (kg)
1-3-11 CX water, L1	677,721	86	0	0	466,818	0	1,528	126,978	0	3,096	1,276,226
1-5-17 CX, fuel, L1	688,023	86	138	0	466,818	0	616	126,978	0	3,096	1,286,666
1-2-8 BF water L1	693,765	86	0	0	466,818	596	616	126,978	99	1,450	1,290,409
1-4-14 BF, fuel, L1	693,765	86	44	298	466,818	596	616	126,978	99	1,450	1,290,751
1-4-20 BF, Direct Delivery,L1	814,272	172	87	694	466,818	595	616	126,978	99	1,450	1,411,811
1-5-23 CX, Direct Delivery,L1	856,207	172	130	0	466,818	0	1,528	126,978	0	3,096	1,454,928

Table 24. cont.

Candidate Architecture	LTV Fuel Consumed (P_{LTV})	LTV Boiloff Losses (B_{LTV})	LTV P/L Boiloff ($B_{P/L}$)	LTV Chilldown Losses ($C_{P/L}$)	CSSV Fuel Consumed (P_{CSSV})	CSSV Chilldown Losses (C_{CSSV})	CSSV Boiloff Losses (B_{CSSV})	MCV Fuel Consumed (P_{MCV})	MCV Chilldown Losses (C_{MCV})	MCV Boiloff Losses (B_{MCV})	Architecture Total Propellant (kg)
He61-2-7 BF water GEO	12,246,445	1,664	0	0	661,374	596	743	102,740	99	1,432	13,015,094
1-5-16 CX, fuel, GEO	12,246,445	1,664	359	0	661,374	0	1,937	102,740	0	2,445	13,016,964
1-4-13 BF, fuel, GEO	12,246,445	1,664	199	1,786	661,374	596	738	102,740	99	1,432	13,017,074
1-3-10 CX water GEO	16,227,386	1,756	0	0	661,374	0	1,937	102,740	0	2,445	16,997,638

Using 30 Layers of Multilayer Insulation (MLI-30)

Thus far, the architecture study has not addressed the mass of the multilayer insulation (MLI). However, multilayer insulation does have mass. Certainly, the mass of 60 layers of such insulation as used throughout this study could be significant.

The first step taken here was to calculate the anticipated boiloff rates for different propellant tanks for various numbers of layers of MLI. The propellant tanks for the CSSV and MCV were chosen. The LEO environment was chosen, since it is the most stressing thermal environment. Boiloff rates were calculated for 10, 20, 30, 40, 50 and 60 layers of insulation for each propellant tank. The mass of the insulation was also calculated. Boiloff masses and insulation masses were compared to see if any relationship could be discerned.

The boiloff rates for the various propellant tanks are shown in Figures 14 -17 below.

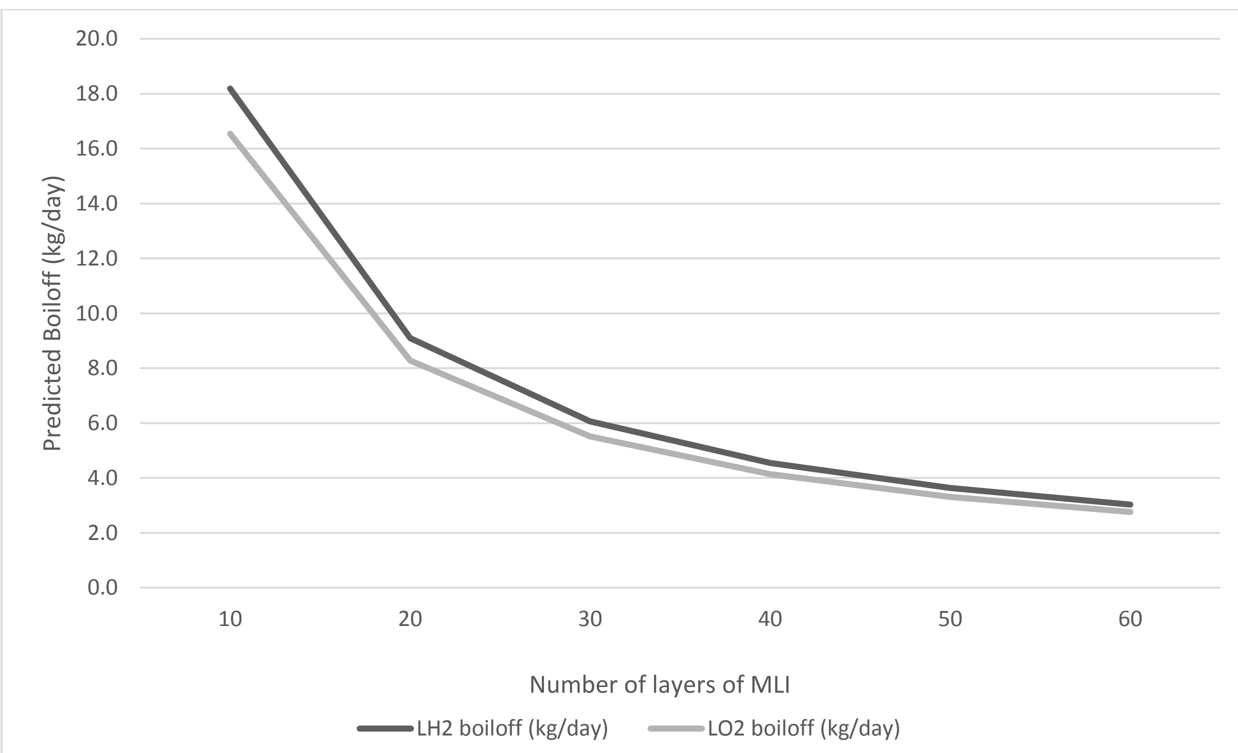

Figure 14. Boiloff Rates for CSSV Bulk Fuel Tanks in LEO.

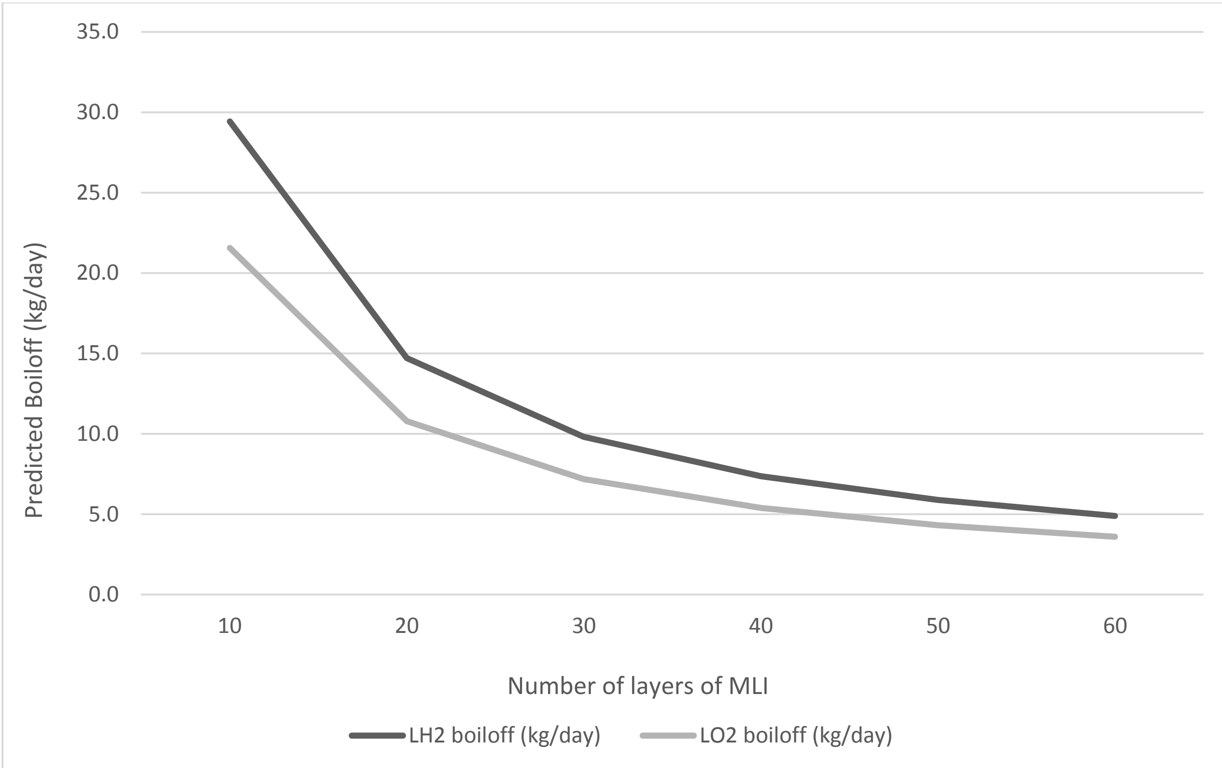

Figure 15. Boiloff Rates for MCV Bulk Fuel Tanks in LEO.

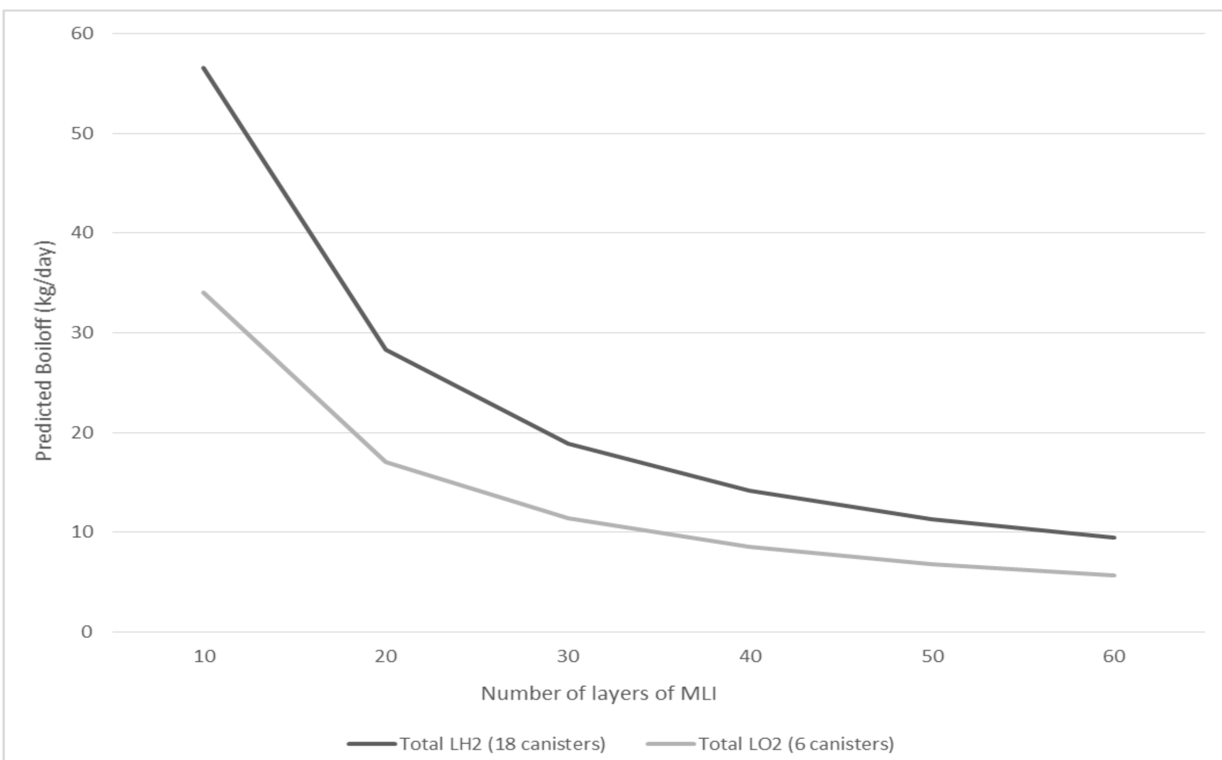

Figure 16. Boiloff Rates for CSSV Canister Fuel Tanks in LEO.

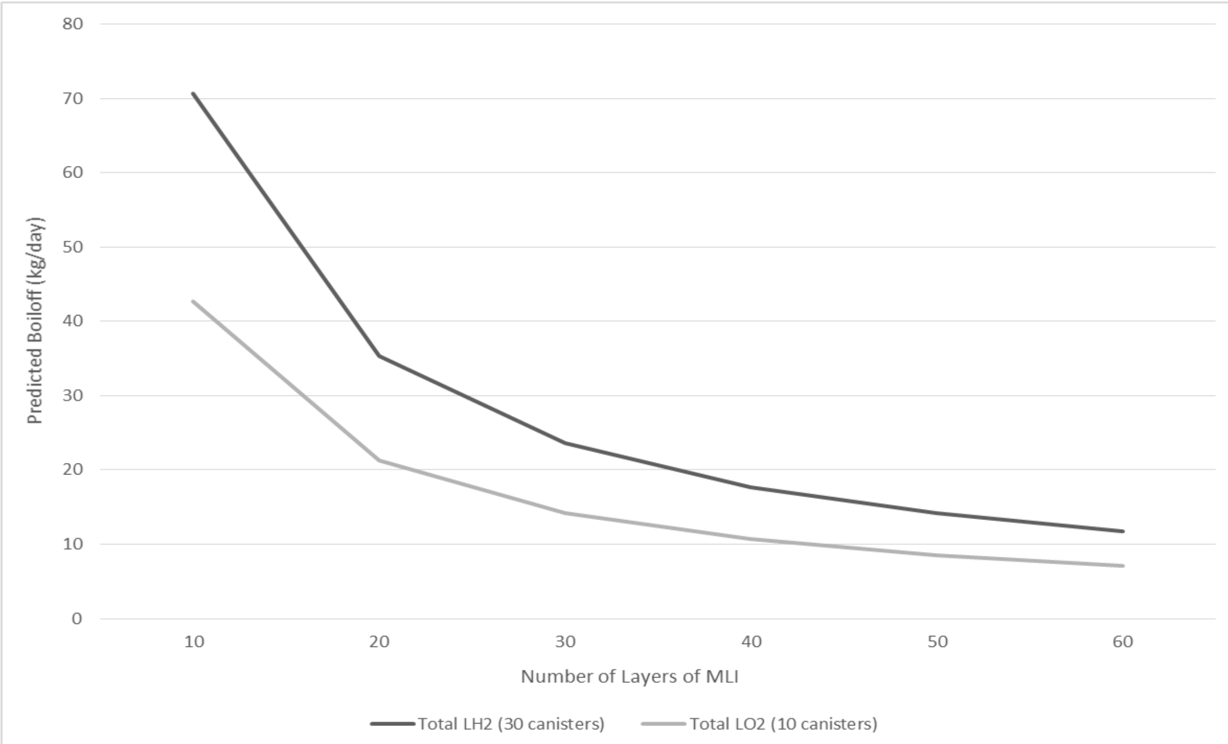

Figure 17. Boiloff Rates for MCV Canister Fuel Tanks in LEO.

In each case, it was noted the increase in predicted boiloff was fairly flat until the number of layers of MLI dropped below 30. Below 30 layers, the curve increases noticeably. Below 20 layers, the predicted boiloff increases sharply. Thirty (30) layers of MLI were chosen for further calculations, and two specific architectures – Architecture 1-4-14 (the delivery of bulk fuel to a depot at L1) and Architecture 1-5-17 (the delivery of propellant in canisters to a depot at L1) – were chosen as the focus of the calculations.

The mass of the MLI was calculated using the surface area of a sphere for CSSV bulk fuel tanks and CSSV/MCV canister fuel tanks. The surface area of the MCV fuel tanks was calculated using the surface area of a cylinder. The density for the layers of MLI was taken from the specifications in the Sheldahl Redbook. A thickness of 2 mils was used .The mass values for the layers of insulation is shown in Table 25 below:

Table 25. MLI Masses for Various Propellant Tanks

Tank	Qty.	Dimension (m)	Surface Area (m^2)	Mass 30 layers (kg)	Mass 60 layers (kg)
CSSV BF LH2	1	3.34m radius sphere	140.2	304	602
CSSV BF LO2	1	2.37 m radius sphere	70.6	153	304
MCV BF LH2	1	10m x 6.41m cyl	358.5	777	1,541
MCV BF LO2	1	10m x 2.29m cyl	229.0	497	984
CSSV CX LH2	18	1.35 m radius sphere	22.9	894	1,772
CSSV CX LO2	6	1.35 m radius sphere	22.9	298	591
MCV CX LH2	30	1.35 m radius sphere	22.9	1490	2,953
MCV CX LO2	10	1.35 m radius sphere	22.9	497	984

Boiloff values for each architecture were assembled for propellant tanks with 60 layers of MLI and 30 layers of MLI. Likewise, the values for the mass of the MLI for each tank with 60 layers and 30 layers was assembled. Delta values for boiloff and mass are shown in Table 26 below. As noted in the table, in order to make a proper comparison, the delta value for the MLI

Table 26. Comparison between Boiloff and MLI Mass Values for 60 and 30 Layers of MLI

Vehicle/ tank	Boiloff - 60 layers (kg)	Boiloff - 30 layers (kg)	Delta (kg)	MLI - 60 layers (kg)	MLI - 30 layers (kg)	Delta (kg)
Architecture 1-4-14 (Delivery of Bulk Fuel to a depot at L1)						
CSSV/BF	1,212	1,828	616	906	457	449[1]
MCV/BF	1,549	2,999	1,450	2,525	1,274	1,251
Architecture 1-5-17 (Delivery of Fuel Canisters to a depot at L1)						
CSSV/CX	1,528	3,055	1,527	2,363	1,192	1,171[1]
MCV/CX	3,096	6,191	3,095	3,937	1,987	1,950
Notes: [1] The delta for MLI mass for the CSSV must be multiplied by six. Each CSSV used for the monthly mission would need the reduced number of layers to achieve the overall boiloff value shown.						

mass must be multiplied by six to account for the six missions in the six month period covered by the objective function.

Architecture 1-4-14:

Increase in boiloff mass = 616 + 1,450 = 2,066 kg

Decrease in MLI mass = (6)(449) + 1,251 = 3,945 kg

2,066 kg increase in boiloff << 3,945 kg decrease in MLI mass

For this architecture, the increase in boiloff mass is much less than the savings in MLI mass, so this suggests that further investigation is needed to determine the "right" amount of MLI to balance expected boiloff with overall spacecraft mass.

Architecture 1-5-17:

Increase in boiloff mass = 1527 + 3,095 = 4,622 kg

Decrease in MLI mass = (6)(1,171) + 1,950 = 8,976 kg

4,622 kg increase in boiloff << 8,976 kg decrease in MLI mass

For this architecture, the increase in boiloff mass is also much less than the savings in MLI mass, so this again suggests that further investigation is needed to determine the "right" amount of MLI to balance expected boiloff with overall spacecraft mass.

The total propellant consumption and losses for architectures 1-4-14 and 1-5-17 were 1,231,104 kg and 1,277,059 kg, respectively. Even with the increase in boiloff, roughly double that for 60 layers of MLI, boiloff accounts for less than one percent of the total consumption and losses for each architecture. Cryogenic propellant boiloff, while a concern, does not appear to be a deciding factor in the choice of architectures.

CHAPTER X

DISCUSSION

Answering the Research Question

The research question asked, "Which architecture satisfies the Design Reference Missions (DRMs) for the least amount of liquid oxygen (LO2) and liquid hydrogen (LH2) consumed in flight or lost due to boiloff?" This question is easily answered at this point: the architecture which satisfies the Design Reference Missions (DRMs) for the least amount of liquid oxygen (LO2) and liquid hydrogen (LH2) consumed in flight or lost due to boiloff is Architecture 1-2-8, in which bulk water is shipped to a depot at L1, where electrolysis and liquefaction would be performed. (Figure 18) This architecture requires less Δv than shipping to a depot in GEO. Shipping in bulk takes advantage of the smaller total surface area of bulk propellant tanks versus that for combined canister tanks. Finally, shipping water to the depot avoids any losses of payload to boiloff during the flight, as well as any chilldown losses between the LTV and the depot.

Understanding the Results

It is difficult to understand "what is going on" with respect to the study results without taking a closer look at what the data might tell us. These data are more readily understandable in Table 27 below. The table displays key information – the architecture number, the location where fuel or water was delivered, the method of transfer, the mass of propellant consumed, lost to boiloff, and lost to chilldown. It is clear that L1 is the least costly orbital depot location to

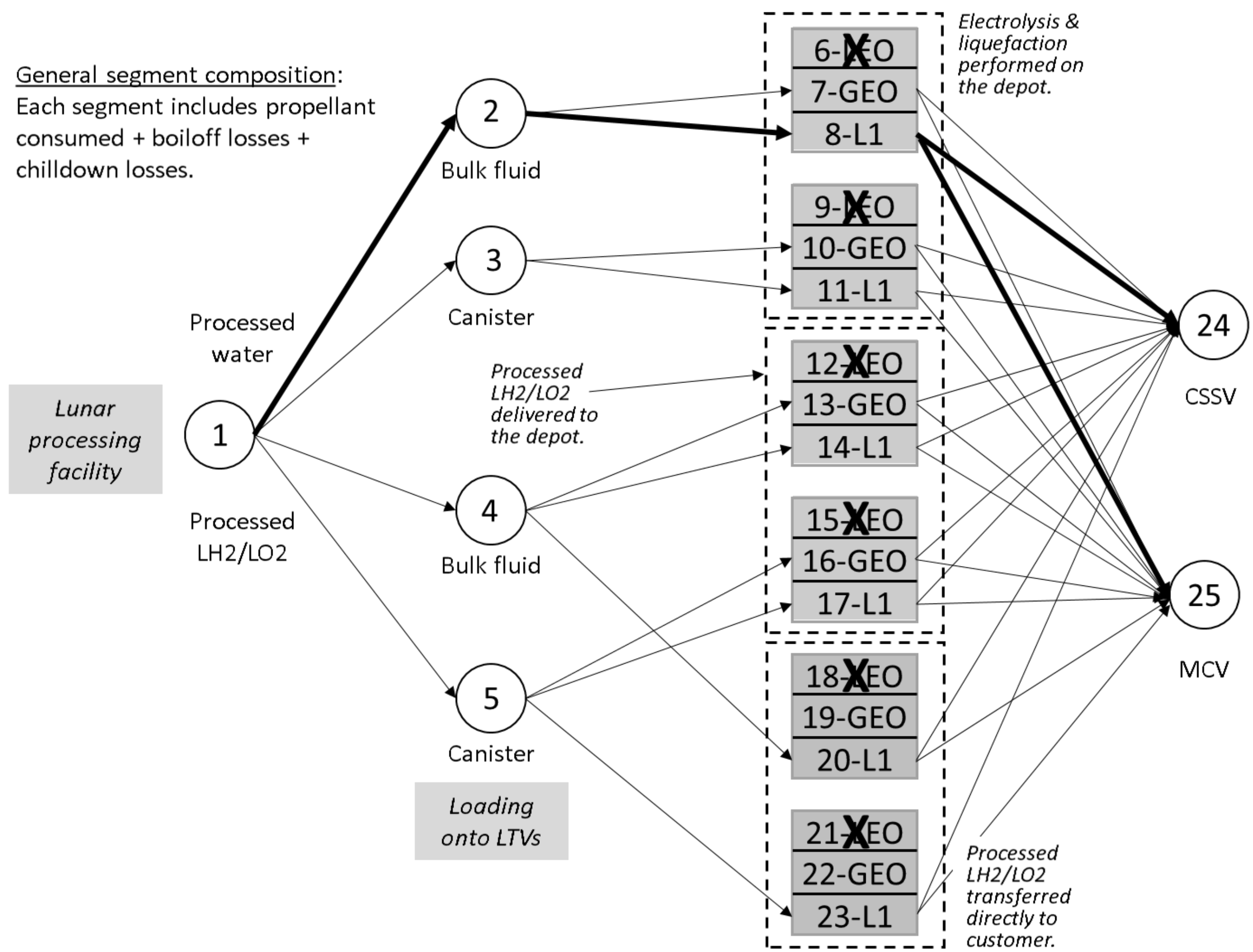

Figure 18. Final Architecture Network Diagram

support, in terms of Δv and overall fuel consumption. For the flights to L1, the bulk fuel method of transfer is preferred. Even though chilldown losses are incurred by this method, the smaller surface area of the bulk fuel tanks results in less boiloff than for canister exchange. However, for deliveries to GEO, the increased boiloff seen with canister exchange is exceeded by the chilldown losses when a large number of LTV flights is involved, each of which involves chilldown losses when transferring from the LTV to the depot.

Table 27. Architecture Study Key Results

Architecture/ LTV Flights	Location of transfer	Method of transfer	Propellant Spent (kg)	Boiloff Losses (kg)	Chilldown Losses (kg)	Propellant Total (kg)
1-2-8/ 6 (water)	L1	BF	1,225,396	2,162	696	1,228,254
1-4-14/ 6	L1	BF	1,225,396	2,217	1,292	1,228,905
1-5-17/ 6	L1	CX	1,267,347	4,857	0	1,272,203
1-3-11/ 6 (water)	L1	CX	1,270,350	4,739	0	1,275,088
1-4-20/ 7 DD	L1	BF	1,308,801	2,236	794	1,311,830
1-5-23/ 7 DD	L1	CX	1,362,925	4,908	0	1,367,832
1-2-7/ 53 (water)	GEO	BF	13,010,559	5,464	696	13,016,719
1-5-16/ 53	GEO	CX	13,010,559	8,023	0	13,018,582
1-4-13/ 53	GEO	BF	13,010,559	5,736	5,953	13,022,248
1-3-10/ 74 (water)	GEO	CX	16,991,500	8,974	0	17,000,474

Other Factors in Selecting the "Best" Architecture

That architecture 1-2-8 is the architecture which used the least resources does not necessarily mean it is the "best" architecture. Shipping water to the depot to be processed into liquid hydrogen and liquid oxygen would require large amounts of electric power on the depot, increasing its complexity and cost.

In the early stages of supplying spacecraft with LH2/LO2 from the Moon, keeping the depot on the Moon (recommended by Oeftering) makes a lot of sense. Tanker vehicles deliver only the mass of propellants needed, and expend only the mass of propellants necessary to deliver them.

As the demand for propellant services expands over time, the establishment of an orbiting depot (as opposed to a depot on the Moon) effectively disengages the schedules of the supplier and customer. We see this routinely on the Earth, where tankers deliver gasoline and diesel fuel to a service station, often at night. Customers for the gasoline and diesel fuel come and go, but their arrival times are not connected to the tanker delivery schedules. There is no rendezvous between the tanker and the individual consumer.

The use of canister exchange, of course, requires the canisters to be handled -- taken from one vehicle and placed on another vehicle. How to accomplish such transfers without damaging the multilayer insulation would be a significant challenge, and was outside the scope of this thesis. Likewise, each canister tank would require connections on the customer vehicle to pipe the propellant to the engine. It is reasonable to assume the number of connections and pipes for a canister arrangement would far exceed those for bulk fuel transfer and add considerable mass to the vehicle.

CHAPTER XII

CONCLUSIONS/RECOMMENDATIONS

Conclusions

Of the potential methods discussed for judging the goodness of candidate architectures, calculating fuel consumption and losses gives the greatest credible insight into potential fuel depot operations. This is because the supporting tools – orbital mechanics, space vehicle design, thermal analysis, boiloff calculation, and others -- are well established.

Earth-Moon L1 is the best location for an orbiting depot; the thesis statement was not supported by the analyses. Because of the reduced Δv requirements, supplying a depot at L1 provides the most fuel for the least cost (in fuel consumption and losses) to transport it.

Low Earth Orbit is not a viable location for a depot supplied from the Moon. The fuel required for the lunar tanker vehicle to fly to a depot or customer in LEO far exceeds the lift capacity of the vehicle. Even if the propellant needed was within the vehicle lift capacity, propellant delivered would be a fraction of what was consumed.

Boiloff would not be the primary factor in choosing among competing architectures. For fuel tanks with 60 layers of MLI, propellant boiloff did not result in crippling losses of propellant, even for the transit to Mars. This suggests that so-called zero boiloff (ZBO) technologies, such as cryocoolers, may not be required for these vehicles. Carrying more fuel is the more simple solution.

The payload capacity of the MCV is not limited by its propellant mass, but by the propellant mass remaining after launch. The fuel remaining after achieving low earth orbit limits the payload mass that can be taken forward to the depot for refueling. If a similar DRM is ever contemplated for a cargo vehicle to Mars, propellants to refuel the vehicle would have to be prepositioned in Low Earth Orbit (like the payload) to maximize the payload mass.

For the propellant tank configurations used, and the fuel transfer pipe dimensions of 10 meters by 0.1 meters, canister fuel tanks appear to offer a competitive alternative to bulk fuel transfers. Although the use of canisters results in increased boiloff compared to the use of larger bulk fuel tanks, the bulk fuel tanks incur chilldown losses which negate their advantages when large numbers of shipments are involved.

The use of canisters often limits the use of the full payload capacity of the host vehicle. This was seen most vividly when shipping water in canisters on the LTV to a depot in GEO. Each LTV could only carry a single canister of water, leaving almost a third of its payload capacity unused, and greatly increasing the number of LTV flights required.

Optimization of the DRM vehicles for their assigned tasks is both possible and necessary. The sensitivity analyses revealed the LTV with 2 engines performed better than the LTV with one engine, and that trade studies are needed to determine the right balance between MLI mass and predicted losses to boiloff.

Recommendations

Recommend revisiting the analysis with better tools to calculate the Δv and time-of-flight values. While the restricted two-body techniques here are a good first approximation, there are software programs on the market (and within the federal Government) that would provide more accurate results. Among others, these include STK (formerly Satellite Tool Kit) and NASA/ Goddard Space Flight Center’s General Mission Analysis Tool (GMAT).

Recommend revisiting the analysis with the mass of the propellant tanks, the mass of the MLI, and the mass of connecting hardware and fuel lines. This information was not addressed in the main portion of this thesis.

Recommend investigating the amount of time spent by the CSSV in the Earth's shadow. An examination of the boiloff results for the CSSV reveals that about 57% of its overall losses were incurred in LEO. Accounting for the time spent in Earth's shadow would significantly reduce that number, and have the effect of "sharpening the pencil" on the study results. For the MCV, its time in LEO (24 hours) is far overshadowed by the number of hours spent in transit to Mars, so the time spent in Earth's shadow is of little consequence.

Recommend examining classic operations research models to see how they can be applied to the depot architecture problem. The vehicle routing problem, traveling salesman problem, and transshipment problem may all have some application. With the architecture study as background information, a survey of these models could be made from a more informed perspective.

APPENDICES

Appendix A
List of Acronyms

AU	Astronomical Unit
BF	Bulk fuel
CX	Canister Exchange
CFM	Cryogenic Fluid Management
CPST	Cryogenic Propellant Storage and Transfer
CSSV	Commercial Satellite Servicing Vehicle
DRM	Design Reference Mission
EDS	Earth Departure Stage
ESAS	[NASA] Exploration Systems Architecture Study
ET	[Space Shuttle] External Tank
GEO	Geostationary Orbit
HT	Hohmann Transfer
ISS	International Space Station
LAD	Liquid Acquisition Device
LEO	Low Earth Orbit
LH2	Liquid hydrogen
LO2	Liquid oxygen
LTV	Lunar Tanker Vehicle
L1	Earth-Moon Lagrange Point #1
L2	Earth-Moon Lagrange Point #2
MCV	Mars Cargo Vehicle
MLI	Multilayer Insulation
NTP	Nuclear Thermal Propulsion

OTV	Orbital Transfer Vehicle
PPLS	Propellant Production and Liquefaction Spacecraft
SOFI	Spray-on Foam Insulation
TRL	Technology Readiness Level
TMI	Trans-Mars-Injection
T/W	Thrust-to-weight ratio
ULA	United Launch Alliance
VD-MLI	Variable Density Multilayer Insulation
ZBO	Zero Boiloff

Appendix B
Register of Ground Rules and Assumptions

- There is no intent to examine lunar mining techniques. These are assumed to be present and mature enough to supply demand.
- Lunar ice deposits are assumed to be large enough to support demand.
- There is no intent to examine economic feasibility of a Moon-supplied fuel depot, or to compare the economics of a Moon-supplied depot with an Earth-supplied depot.
- There is no intent to determine how large the “fleet” of lunar tanker vehicles should be. The size of such a fleet would be driven by the number of supply missions required to service the depot.
- There is unlimited power on the Moon to support mining operations and electrolysis and liquefaction.
- The lunar infrastructure to support mining operations, electrolysis and liquefaction, and LTV launches and landings is already in place.
- LTVs are assumed to launch from the lunar equator.
- The study assumes circular, coplanar orbits for the Earth, Moon, Mars, and candidate depots, and the orbits of these bodies are coplanar with the Sun.
- The study assumes a zero angle between the spacecraft and the earth, and the spacecraft and the Sun; that is, fuel tanks receive full exposure to solar flux, earth reflected heating, and earth infrared radiation.
- Restricted two-body techniques are used to calculated the orbits used in this study. Impulsive maneuvers for all spacecraft are assumed.

- The study assumes “zero boiloff” (ZBO) technology (active cooling) is used on the depot. There is no loss of propellants while at the depot. For those architectures in which electrolysis and liquefaction are performed on the depot, there are no losses of propellants.
- Where water is transported to the depot for processing, boiloff is assumed to be zero.
- All spacecraft (other than an orbiting depot) use passive insulation only.
- The mass of spacecraft propellant tanks is not considered.
- The mass of MLI blankets is not considered.
- The amount of time needed to transfer bulk propellants or to exchange propellant canisters is not considered. The study assumes slow-fill (ventless) transfer of cryogenic propellants.
- No attempt was made to characterize the thermal environment of a conjunction class trajectory to Mars. Instead, the thermal environment calculated for Earth-Moon L1 was used. Since Mars is farther from the Sun than L1, using the values calculated for L1 would represent a worst case scenario. Actual boiloff rates during transit would be less than the values used.
- All operations (orbital maneuvers and fueling operations) are controlled robotically.
- Except for MCV bulk fuel tanks, all other tanks are spherical.
- The study assumes the Mars Cargo Vehicle (MCV) could use canister exchange as a means of fuel transfer. This is incorrect. Since the MCV launches from the Earth, the MCV’s bulk fuel tanks are part of the vehicle’s thrust structure and must withstand launch forces. It would be difficult, if not impossible, to configure canister propellant tanks to withstand such forces. This assumption was made to allow the various candidate architectures to be examined.

Appendix C
Dictionary of Constants Used

Constant	Value	Source
Average albedo (earth)	0.367	Planets and Pluto (2008)
Density of LH2	70.99 kg/m^3 @ 33 psi	NIST
Density of LO2	1191.6 kg/m^3 @ 21 psi	NIST
Density of Water	1000 kg/m^3 @ 40 psi	NIST
Distance from Earth to L1	322,127 km	Sellers (2005)
Distance from Earth to Moon	384,400 km	Sellers (2005)
Distance from L1 to Moon	62,273 km	Sellers (2005)
Earth radius	6,378 km	Wertz & Larson (2010)
Geostationary Orbit (GEO)	42,164 km radius	Sellers (2005)
Surface gravitational acceleration-Earth	9.81 m/s^2	Sellers (2005)
Surface gravitational acceleration-Moon	1.62 m/s^2	Williams (2015)
Heat of vaporization – LH2	448,690 J/kg	Airliquide (2013)
Heat of vaporization – LO2	213,050 J/kg	Airliquide (2013)
I_{sp} for RL-10B2 rocket engine	465.5 seconds	Aerojet-Rocketdyne
I_{sp} for J-2X rocket engine	449 seconds	NASA
Oxidizer-to-Fuel Ratio (Used in tank sizing)	6:1	Huzel & Huang (1992)
Solar Constant	1,367 W/m^2	Wertz & Larson (2010)
Stefan-Boltzmann constant	5.67051×10^{-8} W m^{-2} K^{-4}	Wertz & Larson (2010)

Appendix D
Glossary of Formulas and Variables

Calculating surface temperature of a spacecraft (solve for T)*:

$\sigma T^4 = [(\alpha/\varepsilon)(S) + (\alpha/\varepsilon)(RH) + E] \times (A_p/A)$

where T = spacecraft temperature (K)

σ = Boltzmann's constant = 5.67051×10^{-8} W/m^2 K^{-4}

α = absorptivity *(= 0.14 for outer layer of MLI used here)*

ε = emissivity *(= 0.6 for outer layer of MLI used here)*

S = solar constant (1,367 W/m^2)

RH = Earth reflected heating

E = Earth infrared

A_p = projected area of the propellant tank

A = total surface area of the propellant tank

** Adapted from Wertz, J. and Larson, W. (Eds.) Space Mission Analysis and Design, 3d Ed. New York: Springer, 1999, p.435.*

Modified Lockheed Model* (Calculating total heating rate (W/m^2))

$$q = 0.00024*(0.017+7E\text{-}6(800-T) + 0.0228*\ln(T))*(N*)^{2.63}(T_h-T_c)/N_s$$
$$+ 4.944E\text{-}10*\varepsilon*(T_h^{4.67}-T_c^{4.67})/N_s + 1.46E4*P*(T_h^{0.52}-T_c^{0.52})/N_s$$

where

q = heat transfer rate (W/m^2)

ε = emissivity of the inner layers of MLI (0.035 used here)

T_h = temp on outside tank surface (K)

T_c = propellant temperature (K)

T	=	$(T_h+T_c)/2$
N^*	=	number of layers/cm of MLI
N_s	=	number of layers of MLI, and
P	=	pressure between layers of MLI (Torr)

** NASA/TM –2004–213175: Analytical Modeling and Test Correlation of Variable Density Multilayer Insulation for Cryogenic Storage, p. 25.*

Vis-viva Equation

$v^2 = GM\,(2/r - 1/a)$

where:

v	=	relative speed of the two bodies
r	=	distance between the two bodies
a	=	semi-major axis
G	=	gravitational constant
M	=	mass of the central body

The product of GM can also be expressed using the Greek letter μ.

Rocket equation (Calculating fuel requirements for specific maneuvers):

$\Delta v = I_{sp}\, g_o \ln (m_i/m_f)$

Where I_{sp} = specific impulse (seconds)

g_o = Earth’s surface gravitational acceleration, 9.81 m/s^2

m_i = initial vehicle mass (kg)

m_f = final vehicle mass (kg)

Earth-infrared (Calculating spacecraft thermal environment)

Earth infrared = $\sigma T_e^4 F$, where T = 289 K, and $F = 1/H^2 = (R/r)^2$, where R = the radius of the Earth, and r = the distance from the center of the Earth to the spacecraft.

Earth Reflected Heating (Calculating spacecraft thermal environment)

Earth reflected heating = 1,367 (0.367) (F), where $F = 1/H^2 = (R/r)^2$

where R = the radius of the Earth, and

r = the distance from the center of the Earth to the spacecraft.

Other general formulas

Surface area of a sphere = $4\pi r^2$

Volume of a sphere = $4/3\pi r^3$

"Projected" surface area of a sphere (area of a circle) = πr^2

Volume of a cylinder = $\pi r^2 h$, where h is the length of the cylinder

Surface area of a cylinder = $2 \pi r^2 + 2 \pi r h$

Projected surface area of cylinder (area of a rectangle) = diameter x length

Appendix E
Sample Oxidizer-to-Fuel Calculation

Fuel tanks for rockets are sized for the desired oxidizer-to-fuel ratio, called the O/F ratio. For LO2/LH2 rocket engines, the O/F ratio is typically 5.5-6.0:1 (Huzel and Huang, 1992). If we abandon fixed bulk fuel tanks for standardized canister tanks, it can be shown that using three LH2 tanks for every LO2 tank still maintains an acceptable O/F ratio.

Example:

Using a 6:1 O/F ratio, 140,000 kg of propellant would consist of 120,000 kg of LO2 and 20,000 kg of LH2.

The volume of the LO2 tank can be calculated by dividing the mass of the LO2 by the density of the LO2:

$$120{,}000 \text{ kg} / 1{,}191.6 \text{ kg/m}^3 = 100.7 \text{ m}^3$$

Multiplying this volume times three tanks times the density of LH2 yields the mass of the LH2:

$$100.7 \text{ m}^3 \times 3 \times 70.99 \text{ kg/m}^3 = 21{,}446.1 \text{ kg of LH2}$$

The O/F ratio for the standardized canister tanks then becomes:

$$120{,}000 \text{ kg} / 21{,}446.1 \text{ kg} = 5.595$$

This O/F ratio is within the stated range for LO2/LH2 rocket engines.

Appendix F
LTV Thrust-to-Weight Calculation

The thrust-to-weight ratio (TWR) is the ratio of the thrust of the rocket to the weight that thrust must overcome in order to propel the rocket skyward. The weight, in turn, is the mass of the rocket times the local gravitational constant (Kerbal Space Program, 2015). Thus,

$TWR = F_{thrust}/m \times g$

By knowing the local gravitational constant and the thrust, the maximum mass of the vehicle can be calculated. Then, by subtracting the dry mass of the vehicle, the mass of the propellant and the payload can be determined.

For the Lunar Tanker Vehicle:

- Thrust of the J-2X engine is 1,307 kN (kiloNewtons)
- Gravitational constant for the Moon is 1.62 m/s^2
- Dry mass = 22,470 kg

As noted in Chapter IV, the optimum thrust-to-weight ratio for ascent/descent for lunar vehicle is about one-half Earth's surface gravitational acceleration.

$0.5 \times 9.81\ m/s^2 = 4.905\ m/s^2$

Dividing this value by the surface gravitational acceleration of the Moon yields:

4.905 m/s2 / 1.62 m/s^2 = 3.027, essentially a thrust-to-weight ratio of 3.

Setting the thrust-to-weight ratio at 3 yields:

$TWR = F_{thrust}/m \times g_{Moon}$

$3 = 1{,}307{,}000\ N/\ (m \times 1.62\ m/s^2)$

Solving for m:

$m = 1{,}307{,}000\ N\ /\ (3 \times 1.62)$

$= 268{,}071.22\ kg\ \sim 268{,}071\ kg$

Subtracting the vehicle dry mass yields the mass that can be allocated to vehicle propellant or propellant to be delivered to a customer:

268,071 – 22,470 = 245,601 kg

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