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The Purpose of Human Spaceflight and a Lunar Architecture to Explore the Potential of Resource Utilization

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Our current national, government-funded human spaceflight effort lacks clarity of purpose. Despite NASA's obsession with human missions to Mars, a return to the Moon offers more benefits, a larger number of near-term milestones, and prepares us for future missions to the planets. We offer a logical, justifiable alternative to the existing program that identifies a compelling purpose for human spaceflight – to learn how to use the material and energy resources of space to create new spaceflight capabilities. As in our previous plan, robotic surface elements begin harvesting lunar water for use prior to human arrival on the Moon, resulting in the creation of a permanent, space-based cislunar transportation system. However, we continue this effort with two significant updates. We use the SLS Block-1 and Block 1B configuration launch vehicle for outpost buildup, which allows much more mass and volume in a single launch, and craft an architecture that minimizes cost (almost 50% reduction in crew cycle cost compared to our previous architecture) for a crew lunar mission cycle by relying upon Commercial Crew launch services. Employing a reusable in-space cislunar crew stage and a reusable human lunar lander, a crew can be launched commercially by any of several providers and returned 6-months later using the Commercial Crew service. To improve safety, we station an Orion at a Low Lunar Orbit (LLO) fuel depot that can be used as an assured crew return vehicle at any time in case of emergency. As a consequence of this strategy, we develop more capability to harvest lunar water for propellant compared to the previous architecture; at the end of the 16-year first phase of the architecture, we produce more than 300 metric tons of lunar water per year, with a production capacity of 500 metric tons per year. We use aerobraking during Earth return to recover the reusable cislunar crew stage; this non-propulsive maneuver removes excess energy for an insertion to Low Earth Orbit to transfer crew to the Commercial Crew vehicle before returning home. We take advantage of a LEO fuel depot, loaded by commercial or government water deliveries to the depot from Earth, to fuel the cislunar crew stage on its way to the Moon. The use of both commercial crew and commercial water transferred to the LEO fuel depot allows the campaign to better use and stimulate commercial space industry, transferring technology and experience from NASA to the commercial sector regarding the ability of humans to use local (off-planet) resources in an effective way to explore and grow off-planet. The total estimated cost for this new architecture is $ 87.7 billion, about $ 550 million more than our previous plan. In addition, we have examined possible international contributions to the architecture, with specific suggestions for bartered and in-kind contributions. With these possible contributions, we can reduce the peak NASA funding to $ 5.5 billion per year while reducing the total program cost to $ 69 billion, a reduction of roughly one-quarter (25%). At the end of the first phase of the lunar campaign, we will have demonstrated and determined the degree to which humans can effectively use local resources to live and thrive off-planet. At that point, future missions to other deep-space destinations (like Mars) can be undertaken, leveraging the technology gained and lessons learned from the lunar experience as well as utilize the consumables and propellant produced from lunar resources. These new products can be used and exploited by government, commercial, or international entities as we continue to expand our reach in cislunar space and beyond.

Nomenclature

CCCrew = Commercial Capability Crew
CCS = Cislunar Crew Stage
CEV = Crew Exploration Vehicle (Orion)

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CL = Cargo Lander
CM = Command Module
DMF = Dry Mass Fraction
DV = change in velocity (propulsive)
EH = Excavation/Hauler
EUS = Enhanced Upper Stage
EVA = Extra Vehicular Activity
GH_2 = Gaseous hydrogen
GO_2 = Gaseous oxygen
HL = Human Lander
iCPS = Interim Cryogenic Propulsion Stage
ISECG = International Space Exploration Coordination Group
Isp = specific impulse
ISS = International Space Station
K = Kelvin temperature scale (0 K = absolute zero, -273°C)
kN = kilonewton
kW = kilowatt
L-1 = Earth-Moon Lagranian Point 1
lbf = pounds of thrust
LEO = Low Earth Orbit
LH_2 = Liquid Hydrogen
LLO = Low Lunar Orbit
LOX = Liquid Oxygen
LSC = Lander Support Cart
MPP = Mobile Power Package
MPP = Mobile Power Platform
NASA = National Aeronautics and Space Administration
NRE = Non-Recurring Engineering
PTV = Personnel Transfer Vehicle
R&D = Research and Development
RFT = Rover Fueling Tanker
RHL = Robotic Heavy Lander
RL-10 = cryogenic rocket engine
RLEP = Robotic Lunar Exploration Program
RWTL = Reusable Water Tank Lander
SLS = Space Launch System
SM = Service Module
SSTO = Single-Stage to Orbit
t = metric tonne (1000 kg)
TEI = Trans-Earth Injection
TLI = Trans-Lunar Injection
VSE = Vision for Space Exploration
WEFSP = Water Electrolysis and Fuel Storage Package
WP&SP = Water Processor and Storage Package
WT = Water Tanker

I. Introduction

The human race is remarkably fortunate in having so near at hand a full-sized world with which to experiment: before we aim for the planets, we will have had the chance of perfecting our astronautical techniques on our own satellite…the conquest of the Moon will be the necessary and inevitable prelude to remote and still more ambitious projects." – Arthur C. Clarke, 1951[1]

We have previously presented an architecture that established the infrastructure for routine space travel by taking advantage of the Moon's resources, proximity, and accessibility[2,3]. In the intervening years, there still appears to us to be an uncertainty and disagreement associated with the underlying purpose of human spaceflight (as funded by the government), even though the stated NASA (and administration) "goal" is to ultimately get to Mars[4]. Such a goal seems to lack credible rationale, which can only be justified through honest and open debate on NASA's purpose for human spaceflight, from which any destination-oriented goal like Mars should be derived. We argue that a logical purpose that can be defended to the US taxpayer offers a better path forward for the near term for human space flight. Once a purpose is approved by critical stakeholders, then a logical destination follows. We will offer a more cogent and defendable purpose for human spaceflight and describe the logic that narrows the selection of near-term destination away from Mars and toward the Moon as the best place to begin pursuit of this purpose. Note that a different purpose drove us to the Moon 50 years ago, and now we should go back not because of some unfulfilled promise, but because it best optimizes the chance of a successful implementation of our 21st Century purpose for human spaceflight.

When the Vision for Space Exploration (VSE) was summarily canceled, there was no vetting process for the underlying purpose of NASA human spaceflight[5]. The Flexible Path scenario outlined by the Augustine committee[6] was merely a discussion of what destinations might become possible, not what purpose should drive the decision to choose one or the other. Because this decision is so critical to NASA and our continued space leadership and because we are about to embark on an administration change, now is the perfect time to dialog and interact with a wider audience on the purpose of human spaceflight. Further, because the time epoch of any new human space endeavor will span several administrations, it is critically important to have the discourse with both the incoming administration, NASA itself, and the Congress, to have any hope for any meaningful outcome. Absent consensus on a justifiable reason for the purpose of human spaceflight (which would then lead to a determination of the most practical destination), our U.S. capabilities in space will atrophy, and our leadership will erode until we will no longer be able to claim American leadership in human spaceflight.

One criticism of NASA's human space flight goal of going to Mars is that it is essentially an abdication of a meaningful decision or commitment for human spaceflight to a future administration, since most of the funding associated with executing a Mars mission will need to occur in future administrations, not the present one. Hence, sound-bite Mars proclamations cost very little now, with promises of vague future commitments that don't have to be debated until then. True, some studies are on-going, and some key initial elements are being developed now (more because of insistence by Congress than any significant leadership from NASA or the administration). However, the large budgetary outlays are all out in the future. *As long as a space goal remains on the horizon outside of a current administration, promises of future funding are simply a cost-free statement of desire, not commitment*. A much more effective approach for the U.S. would be to elevate the dialog regarding the future purpose of human spaceflight to both the administration and Congress, so that in the end (assuming that any future endeavor can be agreed to) there is commitment from both the funding body and the implementing body to better ensure that the long term endeavor comes to fruition.

There has been some progress on certain capabilities being developed within NASA that we believe would be worthy of consideration for another instantiation of our architecture. Two such elements have caused us to recast the architecture to take advantage of these new elements so that we enhance that success probability, reliability, flexibility, and future potential of the lunar campaign while preserving the total cost that we projected earlier.

The first of these elements is the Commercial Crew Program. If successful, NASA will have the option to buy crew seats to Low Earth Orbit (LEO) from a commercial entity. While this capability is still in development, it is planned to be in place in a few years to support ISS operations. As such, we can leverage this capability in our architecture by crafting cislunar elements that "pick up" the crew from LEO with the same capabilities that Commercial Crew will soon provide and ferry them to the Moon and then back to LEO once the crew mission has completed. This also allows redundancy for a single commercial launch failure, so that the lunar campaign can continue immediately even with a failure. Previously, crew rotations in our architecture were performed by NASA using a heavy lift vehicle, which would temporarily end human tending of the outpost (robotic operations could continue) should a failure in the launch vehicle occur. To ferry the crew to the Moon and back from the LEO Commercial Crew handoff, we develop a reusable crew cislunar transfer stage that takes advantage of the latest aeroassist technology to return to LEO and rendezvous with the Commercial Crew element via an orbital depot. Another advantage for use of Commercial Crew is that it opens future avenues to allow outside participation with humans in going to the Moon without a large investment. If NASA funded 2 crew rotations per year, then at some point (provided that lunar fuel would be available) a commercial or international entity could add human flights to LEO with a barter arrangement from NASA to use its cislunar crew vehicle and reusable human lander to get additional crew to the Moon and back.

To compensate for the need of increased lunar water production for this version of the cislunar transportation architecture, we also take advantage of the new Space Launch System (SLS), which has lift capability to Trans-Lunar Injection (TLI) orbit of 25 t for SLS Block 1, and 35 – 40 t for SLS Block 1B, which is greater than we offered in our version of the heavy lift launch vehicle that we assumed previously. This allows us to launch packages of lunar surface systems necessary to increase lunar water production (by a factor of 6 or more) and allows additional unallocated payload mass for use by international partners or commercial entities. Since the development cost is assumed to already be incurred, adding SLS to the architecture greatly enhances the likelihood of success and allows us to add some margin and robustness to other architectural elements. Because our architecture would not carry the SLS development cost burden, it frees up resources to allow for that robustness, including use of Orion as a lifeboat at the Low Lunar Orbit (LLO) Depot, giving emergency return coverage that allows crew to return from the Moon in an emergency.

II. The Near-term Purpose of Human Spaceflight

One of the most vexing challenges facing NASA and the U.S. at the time we wrote the first paper on lunar architecture[2,3] was the need to formulate and articulate a viable purpose to human space flight, as executed by a government-funded space agency. In the intervening time period, this challenge has not been answered and is even more of an issue today, since significant resources have been spent in the meantime without a consensus answer to this question[7]. True, there is continued productive activity at the ISS. However, even in LEO, NASA has been playing more of a caretaker role in executing existing activities and plans for ISS without a firm, realistic long-term purpose or goal in mind for human spaceflight. When asked, the purpose seems to be wrapped around the concept of "exploration." As we have mentioned in the past[3], this term has a number of meanings to different people, which has caused confusion and frustration in many by trying to interpret a different meaning to the same term by different people. Instead, we offer a different perspective in which to debate the purpose of human spaceflight without use of the term.

Fundamentally, NASA is a government engineering research and development organization, established to enhance the country's understanding, capabilities, and interest in space. Focusing on this and extrapolating to the future, we recognize that the adventurous nature of the human spirit will inevitably drive our species off-planet. It seems to flow from these two concepts that NASA's object should be to simply (and affordably) enable the technologies and capabilities of space travel, for both machines and people, so that eventually humans can effectively extend their reach off-planet. There will come a time when industry and commercial progression is strong enough to pick up from the R&D investment that NASA has already made in space for this particular pursuit, but such a time has not yet arrived. This conclusion does not diminish the current growth in the commercial launch industry, but the next steps needed to address human habitation on another planetary body are too new and expensive to be able to rely solely on the investment of private industry. We mention habitation on a planetary body for a reason: the chain that still binds us to Earth is the complete lifeline of materials from Earth needed to sustain human presence in orbit. Only when we break that chain, and begin to use off-world resources, can we take the next step in the journey.

Recognizing this lack of vision for human spaceflight, we propose a primary purpose that we feel can be justified to the U.S. taxpayer as a viable investment with possible significant future return. That is to say, the primary purpose of NASA human spaceflight should be to determine the degree to which humans can reliably and affordably live and work off-planet while enhancing the overall benefit to the country and world (assuming international partnerships). To be clear, living off-planet in this context implies the use of the resources of space to the maximum extent practical and to allow extrapolation of future potential as the government determines how to transition its involvement and investment to a preferred growing and healthy competitive commercial sector or private enterprise. We feel that it is the duty of government to nurture fledgling industries and activities that have a reasonable probability of significant payoff or benefit to the country as a whole. Our proposed purpose is no different than a specific instance of this broader government R&D context.

While the ISS has been a cornerstone for human spaceflight for more than 20 years, our vision looks beyond ISS by adding the dimension of space resource utilization into the mission sphere, which we believe is the next logical step in outward progression of humans in space and the biggest enhancement factor in generating tangible wealth and well-being as a result of human space activity. Absent any significant breakthrough in launch technology away from chemical propulsion to greatly lower the cost to get to space, any affordable human campaign in space beyond LEO (other than "fly-by" human missions to simply stop and look around and then come back) must have a high percentage of materiel closure using local space resources to break the chain of Earth-based logistical support.

For human space activity thus far, we have no real data (but much speculation, analysis, and ground simulation) on the added dimension of use of local resources, and the enabling and enhancing potential of such a shift in thinking to know for sure whether it will practically work or not. We feel that there is enough potential for a huge enhancement factor in using local space resources for human space activities for the creation of wealth and well-being to entertain this concept and ascribe it to human spaceflight. By transitioning from an Earth-centric spaceflight template to one based on the use of space resources of materials and energy, we remove restrictions on mass and power that have kept us capability-limited since human spaceflight began. In short, our goal is nothing less than to become space faring – to have the ability to go anywhere, at any time to achieve whatever goals we imagine to be valuable or useful.

While this approach sounds promising, we must recognize that the next layer is one that is fraught with danger. The implementation of a resource utilization architecture can have the greatest impact on success or failure – if not done properly, it can add cost, time, and skew implementation from being aligned with the mission objectives. We propose a set of practical, measurable objectives (in the MISSION OBJECTIVES Section) associated with this primary purpose that provides the best balance between cost and benefit while setting realistic probabilities of success. We then lay out a modified architecture and associated cost, updated with new capabilities not envisioned earlier, that seems reasonable in context with the overall historical NASA program architectures. By developing a set of Mission Objectives and an architecture in this way, we ensure that the effort is not open-looped, and is bound by measurable objectives that allow us to change direction based upon feedback as we progress toward those objectives. From here, a detailed set of top level, quantifiable requirements can be developed to meet those objectives should this architecture be pursued.

At this point, it is critically important to talk about a realistic (as opposed to an optimistic) cost. Our intent is "not" to offer a cheap buy-in cost to get the program started and then the realistic costs "play out" afterwards and the cost skyrockets. We have done our best to accurately portray the cost for this architecture. This cost will be in the range of several tens of billions of dollars over a few tens of years. Note however, that time is the free variable; that is, as long as the estimates of cost for the individual elements are reasonable and more or less bounding, then the time it takes is simply a matter of the funding available from year to year. Such an approach permits greater flexibility in program execution, a critical requirement given the likely stringent budgetary constraints that future space efforts will face.

III. Destination

Consistent with our purposed purpose of human spaceflight, and taking a fresh look at the determination of destination, the next question to be answered prior to setting objectives and then crafting an architecture is to decide where to go. We maintain that a single destination as opposed to several destinations best serves the practical evaluation of this human spaceflight purpose due to the need to aggregate materiel resources at one location so as to minimize overall cost to the endeavor. Since many elements, components, and systems will be needed to examine the local resource potential, and each of these must eventually come together to support human space activity, it is logical that this be localized at one location.

One of the most useful substances to enable greater spaceflight capability is water. Water can support human life (as both a consumable and as radiation protection), serve as a medium for energy storage (in rechargeable fuel cells), and when broken into its component elements hydrogen and oxygen, it is the most powerful chemical rocket propellant known. Finding accessible and usable deposits of water in space will create an enormous leveraging capability by supplying future human voyagers with a resource of significant utility that does not have to be launched from Earth.

To learn how to use space resources, there are three main possible destinations: Mars, Asteroids, and the Moon. These are chosen as opposed to locations in space because they have material resources and usable mass is needed to pursue our proposed human spaceflight purpose. Physics is unforgiving; transport of mass through space is too expensive to have a staging ground that is not located at a massive object (at least for the near-term). In formulating an architecture to execute any objectives derived from this human space flight purpose, it will take many launches (from a few tens to several tens) to fully execute any mission design that vets our human spaceflight purpose. Almost by inspection, asteroids are ruled out because repetitive and routine access is not possible due to their complicated and difficult orbital mechanics. In addition, although many asteroids contain water, it is not immediately usable in its native form (chemically bound in mineral structures) and requires significant mechanical and chemical processing to extract and purify.

That leaves two destinations, Mars and the Moon. Much has been written about the two destinations. Mars garners favor at the moment as the destination for human spaceflight for the current administration, but for different reasons and purposes than we propose. With our proposed human spaceflight purpose, a review of the two options points clearly to the Moon as the best starting point. The Moon is greatly favored because of its proximity, its resource content and its low gravity. The Moon is "much" closer than Mars, with constant trip opportunities, as opposed to launch windows every 26 months for Mars. Transit time is also a huge factor against Mars, as the long trip time will by necessity require much more transit mass and therefore add much more cost. Just as a thought example, one would need every system on either Mars or the Moon, and then additionally, the Mars journey would need enough robust systems and supplies for the long journey out and back. This cost difference is more than a few percent; it is measured by integer factors. Both the Moon and Mars contain significant quantities of water, the key resource needed to create new spaceflight capability.

A key virtue of the Moon as first destination, its proximity, is critical in that it enables our architecture. We emplace an outpost on the Moon and begin harvesting and processing its resources using teleoperated robotic machines[2,3]. These robots are directly controlled by human operators on the Earth and such control requires near-instantaneous reaction and response, which requires proximity. Such a mode of operations is possible on the Moon (3 light-seconds round trip), but it not possible on asteroids or Mars, where reaction times vary from minutes to tens of minutes. This factor allows much of the outpost infrastructure and refueling depots to be put into place and made operable before people arrive, making our architecture affordable, scalable and safe.

The Moon is small enough to enable one of the most critical affordable architecture decisions to be made, that of deploying a fully reusable single stage lunar lander (lunar orbit to lunar surface and back). One key to affordability lies in the common sense concept of reuse wherever possible. We will show that this element – as well as the reusable crew element to ferry humans between low lunar orbit and low Earth orbit and back – is crucial to develop an architecture that is largely otherwise unaffordable (e.g., big-cost throwaway elements would be required every time there is a crew change). Conversely, there has been no recognized design or concept yet offered for a fully reusable Mars single stage lander (ascent and descent) due to the higher gravity of Mars and the subsequent difficulty of descending to and launching from the surface (Mars surface to Mars Orbit and back) through an atmosphere even as thin as it is on Mars. It is unlikely in the near term to ever be able to design a fully reusable Mars single stage lander.

One more point to be made for the difference in destination times for the two locations: we recognize that there is a large set of unknowns with this activity. Most probable outcomes will require a reasonable response feedback time from Earth with unplanned human missions and/or cargo to compensate for unknowns. Unanticipated critical and potentially fatal problems that occur will require action from Earth (not just knowledge transfer, but transfer of mass from Earth) to fix or ameliorate those problems in the field. With Mars so far away, it is simply untenable to assume that there will be adequate response times requiring Earth action and hardware to solve field problems. We cannot guarantee that such critical problems will occur, but an inspection of past programs suggests that it is prudent to expect them.

While we firmly believe that a single destination at the Moon is the best initial location for exercising this objective, it does not rule out human missions to different destinations for other reasons, purposes, and times. Further, it does not rule out the eventual desire or need at some point in the future to progress to Mars or beyond, but only after we have examined the degree to which local resources can close a human architecture. This lunar campaign should give us enough data to extrapolate into the future for different destinations like Mars, as well as to determine the state of a (hopefully) growing private industry and activity (not to mention international capability) to continue to grow this field of endeavor and shoulder more of the cost burden of a human Mars campaign. The resource extraction and utilization process will probably be different at the two destinations, but there are also commonalities (e.g., the handling and processing of large quantities of granular materials). Enough information will be gathered on the Moon to allow an extrapolation into what can be practically done and what cannot based upon what has been done by that point in time.

IV. Science Symbiosis

Throughout the discussion so far, we have not yet mentioned science. We recognize that there are valid scientific investigations in any destination that we choose. Science does not and should not play into the choice of destination because of the unique purpose that we have offered for human spaceflight. It is more appropriate to consider science in the context of a symbiotic relationship with any human space activity relevant to this topic. Surely, the science of lunar geology and material processing will play a key role in the initial activity at the Moon in pursuit of this human spaceflight purpose. As well, there will be ample opportunity for science to act as a full partner in the overall architecture, with assigned payload masses to be negotiated along with all of the other needs at the time. It is expected that the science, as with all science at NASA, will be competed to ensure the best value.

In this way, we have removed the argument of which destination has the best science. For one, it is very difficult to compare the value of different science investigations. Comparing the science versus the destination for human spaceflight also skews the discussion as to what the best purpose for human spaceflight should be. In our context, science is considered at every major step, but must be consistent with the Mission Goals and Objectives set forth at program inception. For this paper, we offer a cogent, defendable purpose, followed by a logical choice of destination, with a proposed set of Goals and Objectives discussed as part of this paper to accurately decompose that purpose. There are always opportunities for science as a part of any program of spaceflight beyond LEO.

V. Stakeholder Participation

We recognize that this endeavor is a major undertaking, spanning multiple Presidential administrations. The only way to maintain continuity is to partner with both a favorable administration and a favorable Congress, with the goal to have long-term commitments from an administration and Congress in the purpose of human spaceflight, the initial destination, and the overall architectural strategy as codified in a set of objectives that can be agreed-upon by important stakeholders. Congress and the various administrations that span this campaign are as critical a piece in this overall human spaceflight mosaic as NASA is, and must be full participants in the strategic formulation and periodic status of the execution of the effort. In this spirit, we have updated the architecture to take advantage of the existing SLS program and its capability, which has been heavily supported by Congress even when there has been seemingly no convincing purpose offered for human spaceflight beyond LEO.

Additionally, to foster a spirit of international cooperation leveraging from the ISS experience, and also to reduce the cost burden to the taxpayer in search of affordable architecture solutions, we propose that some architectural elements would be excellent candidates for international contribution. There is clearly substantial international interest in returning humans to the Moon[8], and certain architectural elements might lend themselves well toward this goal. At this point in the process, everything should be "on the table" to negotiate initially in the spirit of international teamwork and to see where the best fits would be. We do offer some suggestions that we feel are good candidates for this discussion. Later, we even estimate the cost values for these elements, although a much more thorough cost analysis would need to be performed should we decide to implement this architecture.

Additionally, we want to take advantage of the commercial potential that can be applied to this endeavor, with the understanding that initially it is quite difficult to embrace future commercial capabilities when there is no proven existing capability to reduce risk to an acceptable level. Where there is proven capability, we suggest an approach that capitalizes on that capability. For instance, we anticipate using Commercial Crew to ferry crew to and from the LEO departure point for cislunar transport. We would pay a negotiated fixed amount per crew rotation to transport 4 crew members to LEO for rendezvous and then transport them back to Earth. Another commercial capability that we plan to tap is the delivery of liquid water to a fuel depot in LEO. This is similar to our previous architecture in that we pay a commercial entity a fixed amount per kilogram of water ingested into the depot, at least in early stages of the architecture before lunar water can be delivered back to LEO.

There are many other plans, blogposts, and discussions related to what the commercial sector "could" do for such a campaign, but until such capabilities have been proven, it is unlikely that NASA can rely upon anything other than competitive procurements with expected government insight/oversight. Even then, we would propose a lighter touch in the oversight function, limiting the ratio of government personnel to the contracted work so that only the truly critical things would need to be monitored and approved. Our approach in the ongoing evolution of this lunar architecture is such that the program planners should be closely coupled to the latest commercial activities and be able to recognize when it makes fiscal sense to splice an emerging commercial capability (proven, not speculated) into the architecture provided the risk is low enough. Several examples come to mind, from the government buying water ice ore on the Moon, to buying liquid water on the lunar surface, to renting space in pressurized habitats on the surface of the Moon, to ferrying water up to the fuel depot in LLO.

We feel that the development of the Moon and the opening of cislunar space is too important and too large an endeavor to go it alone, especially since the U.S. likes to talk about U.S. leadership in space. In that light, it is absolutely recognizable that we would be the world leader in space should we undertake the organization and management of an international and commercial team going to develop the Moon.

VI. Architecture Objectives and Implementation

For this architecture, it is important to sustain commitment for a long period of time. We propose to establish with the stakeholders an overarching set of objectives determined from the single goal to "Expand human reach beyond Earth, while opening the human economic sphere using off-planet resources." The intent of creating such a set of Objectives is to reach agreement beforehand with the stakeholders to provide total transparency with them. In addition, we recommend that NASA periodically evaluate the status of progress against each objective to the stakeholders so that they stay engaged throughout the process. Too many times, a long-term endeavor like this is subject to the whims of a small set of decision makers for various reasons seemingly not related to the project in question. The Objectives of this architecture are presented in Table 1.

Table 1. Objectives

Objective Number	Objective Statement	Objective Rationale
		Phase 1 (Years 1 through 15)
1	Demonstrate basic ISRU (ice ore) mining viability and water extraction and long term storage	*The key to this architecture hinges upon the ability to leverage lunar resources for propellant throughout cislunar space, including the lunar surface. This is critically important because the viability of any long term lunar activity requires use of reusable hardware elements and supply from off-Earth sources. Use of ice deposits at the lunar poles is the simplest and easiest way to accomplish this objective, which also enables a viable, affordable single-stage lander transportation architecture due to the high propulsion efficiency (Isp) of hydrogen and oxygen and the low gravity of the Moon. To satisfy this objective, we must show that the ice is accessible and of sufficient concentration to be useful to manufacture propellant. While it is now known that the lunar poles contain water, we do not know its physical form, concentrations, distributions and its difficulty of extraction. This will be an early and critical challenge.*
2	Demonstrate lunar propellant generation and long term storage	*In addition to being able to mine the ice ore, the architecture needs to show that it can successfully process the water from the ore, liquefy and store it with minimal loss. This latter function is usually a significant challenge, but the local topography can be used (small crater interiors) or altered (a perimeter berm) to maintain a cold environment that should help reduce the liquid hydrogen boil-off losses to close to zero. The architecture will need to show that enough continuous power can be provided almost continuously to convert and store propellants with minimal loss.*
3	Demonstrate feasibility of using a reusable single stage to orbit element to provide water to a lunar orbiting asset.	*It must be demonstrated that lunar fuel can be used at a lunar orbiting element to refuel in lunar orbit for the Earth return. This is an outcome based upon the fact that it is not currently possible to launch from the Moon with a SSTO element and then make the journey all the way to the Earth. In addition, it is especially important from the Moon to the LLO exchange point (i.e., a reusable Human Lander), and from the LLO exchange point to LEO (or direct to Earth return). An earlier version of this architecture capitalized on use of a heavy lift vehicle to transport a capsule to LLO, and that would meet the intent of this requirement as long as the return leg was performed using lunar water electrolyzed at an LLO Propellant Depot.*
4	Demonstrate feasibility of electrolyzing lunar-provided water and storing the resulting propellant with minimal loss for the lunar orbiting asset.	*The LLO Propellant Depot must be capable of electrolyzing the water in an efficient manner and storing it with essentially zero boil-off (no losses) for a long period of time (1-2% over several months timescale). This capability must be demonstrated as a prerequisite for determining overall viability of this concept.*
5	Demonstrate feasibility of a reliable, reusable cis-lunar human transportation element using aerobraking on the Earth return leg, fueled for Earth return via lunar propellants.	*The final piece in the architecture is the reusable crew orbit transfer stage. As part of the viability, it is critical to show that this can be safely performed using lunar propellants for the return leg. This step is proof of the viability of the concept of lunar water-to-propellant, since in order to fuel this element for Earth return, it requires all other pieces of the lunar propellant generation capability to succeed.*
6	Demonstrate safe, reliable, and repeatable robotic transport and human crew rotation to and from the Moon using lunar resources to return.	*While all of the individual Objectives may be met above, it is also necessary to show an architecture that can sustain reliable human operation and crew rotation without need for new elements of human transportation shipped from Earth. Sustained reusability is especially important from the Moon to the LLO exchange point (i.e., reusable Human Lander), and from the LLO exchange point to LEO (or direct to Earth return). Consistent, timely, successful crew rotations with effective hand-offs and re-fueling at various points along the way will act as a validation of the entire concept of using planetary resources to live off-planet by first enabling the human transportation component of expanding life off-planet.*
7	Demonstrate viable human habitability on the lunar surface	*While Apollo demonstrated brief periods of human presence at equatorial latitudes on the Moon, it must be shown that humans can reasonably live and work for extended periods at the outpost near the lunar poles. Of particular concern is lunar dust, which is a significant and potential danger for skin irritation and inhalation; this dust can be very difficult to control. These effects will need to be carefully mitigated. Other aspects of long-term life have largely been demonstrated on the ISS program and any unique features associated with the partial gravity of lunar surface residence will need to be examined and confirmed as well as effectively handled.*

Table 1. Objectives (continued)

Objective Number	Objective Statement	Objective Rationale
8	Demonstrate reliable human outpost infrastructure with human maintenance and repair paradigms	*One of the biggest challenges to this architecture is the reliability of the outpost infrastructure in a severe environment. The coldest environments (i.e., a local rover or propellant storage environment) can have temperatures as low as 20 K, while illuminated surfaces normal to the sun vector might get a few hundred degrees Centigrade or hotter for some period of time. In steady state, extreme temperatures are not normally an issue, since thermal designs can be cold or warm biased. The real challenge is with equipment that sees both temperature extremes in their lifetimes, some for many cycles. In addition, the regolith fines will present a considerable problem if not carefully mitigated for all moving parts as well as for human interface activities like egress and ingress or EVA suit operations. It is expected that a focused and disciplined approach to human protection as well as periodic preventive as well as unplanned corrective maintenance for moving and rotating equipment will be a significant part of crew activity. This requirement demands a balanced workload with other activities at the outpost and preserves the outpost performance.*
	Phase 2 (Years 15+)	
9	Demonstrate construction of lunar structures designed for human pressurized utilization and habitation using lunar resources	*While this architecture will have already demonstrated the lunar ISRU water cycle chain, another important lunar commodity to explore and exploit will be use of lunar materials to build pressurized structures. It is recognized that interior parts of such a structure likely would be from the Earth initially, but all primary structure and most secondary structure should be fabricated using local lunar materials. It is important to ensure that the implementation of this Objective is not a token qualitative response but a real and lasting use of lunar materials and corresponding reduction in Earth materials or support hardware for primary and secondary structures. Metrics can be developed (e.g., 100% primary structure, 90% secondary structure, etc.) that will document compliance with this Objective.*
10	Demonstrate production and fabrication of tools, and other structural items necessary for a viable human Outpost using lunar materials	*Similar to the development of pressurized volume, it is necessary to show that local lunar materials can be used to fabricate special purpose items ordinarily delivered from Earth that are required to make things work smoothly at any type of outpost. In directly decomposing the primary Goal, it must be shown that all of the "extras" can be successfully added to the lunar Outpost inventory using materials that do not come from Earth. Naturally, this does not apply to avionics or specialty items, but clearly the structural elements associated with supporting an outpost (tools, slings, ladders, extension devices, even mechanical fasteners, etc) must be produced on the lunar surface. This effort involves the development of metals or ceramic processing and fabrication using lunar resources, and any packaging required to use such devices. As with the previous objective, metrics can be developed to make this Objective meaningful.*
11	Demonstrate the viability of plant (and animal) growth on the lunar surface to support and sustain human Outpost activity.	*As the outpost becomes more mature, the longer term viability targets will need to be addressed, like food production without relying upon an extensive Earth resupply chain for the bulk of the foodstuffs. This will first require additional habitable pressurized volumes, and will also require water, light and agricultural chemicals (e.g., fertilizer) to operate effectively. This objective will examine basic growth potential for various items and the effort required to initiate and sustain such a production. Clearly, there is much science that can and will be done to examine low gravity effects upon plants and animals that may only be peripherally involved in supporting the Outpost; the science associated with these investigations can be performed as Secondary Objectives separate from this particular Objective.*
12	Demonstrate resource efficiency improvement of water (or propellant) upload off of the lunar surface	*Phase 1 of the architecture provides a viable initial step toward the overall primary Goal of extending human reach beyond Earth. To increase sustainability, the architecture should examine ways to upload water more efficiently without the large propellant resource penalty of launching off the Moon to an orbiting Propellant Depot in LLO or beyond. One means that might be explored is to use the concept of a Mass Driver (i.e., electromagnetic launch) on the Moon to inject enough energy into a lunar payload to reach LLO or beyond without loss of propellant. This may take time to develop, but would provide an even bigger payoff, since most of the propellant used for crew rotation in Phase 1 is actually used for the Reusable Water Tank Lander to send water as a payload to the LLO Propellant Depot.*
13	Demonstrate reusable lunar up mass launch capability for goods produced on the Moon	*This Objective examines the portability of goods produced on a planetary body being used away from the surface of that body. The basic objective would be to convert a lander (such as the Cargo Lander) into a reusable lander that can launch off of the lunar surface using lunar propellant with significant payload capability (5 to 10mT) that can be used either in LLO or further transported anywhere in the cislunar system or even beyond. Conceptually, the energy penalty (using chemical propellants) for lifting material off-planet on the Moon compared to Earth is much less, but has never been practically demonstrated and is not without significant technical challenges.*
14	Demonstrate the linkage and growth in off-planet commercial activity as a result of undertaking this program.	*When it is all said and done, the real result of this investment will be its positive influence on non-government space activity in order to grow essentially a fundamental new space industry from its infancy. This investment will have been judged a success if and only if there has been germination and growth of the concept in the commercial sphere, with significant non-government customers. If this happens, then the investment will have been judged as ultimately successful, creating wealth in society as a direct result of government investment in this paradigm-changing concept.*

As mentioned previously, the two major changes to our previously published architecture[2,3] are the use of the Commercial Crew Program to minimize recurring cost for a crew rotation and add launch flexibility in case of a launch failure, and the use of SLS as a cargo launch vehicle to improve upload mass efficiency. First, however, we will review the overall architectural strategy that we use for the lunar campaign.

The overall strategy for lunar outpost architecture must consider the need and objective to seek lunar resources per the Mission Objectives and design to explore the feasibility and availability of processed lunar water in particular to use as propellants for the crew return leg from the Moon to the Earth. Therefore, we have crafted a crew return transportation concept that uses lunar water (converted into propellant) for the Earth return leg. Because the program must be affordable, we have developed an architecture that has key transportation elements that are reusable, with in-situ maintenance and repair capability. We use a reusable single-stage-to-orbit (SSTO) Human Lander (HL) to transport crew from the surface of the Moon to the crew lunar transfer point and then back to the surface as a key element in crew rotation.

To set the location of the lunar transfer point, and recognizing the need to use lunar-processed propellants for the return leg to the Earth, we have set the HL transportation point at Low Lunar Orbit (LLO) at roughly 100 km above the surface as a good location to perform crew transfers from the HL. Since it will be required to use a propellant depot located logically at the transfer point, and because that propellant depot will be loaded by lunar material (either water or LOX and LH_2), and since the need to transport those materials to the depot from the Moon would also be a challenge for a reusable SSTO element, we wanted the transfer point to be as practically close to the Moon as possible so that the design and capability burdens on the lunar landing elements are achievable and minimized. In looking at the energy required for both locations, a SSTO HL would need roughly a little more than 2100 m/s change in velocity (Delta Velocity, or DV) per leg to LLO (including attitude control, rendezvous, proximity ops, and margin), versus 2700 m/s per leg to Earth-Moon L-1. That extra DV (1200 m/s round trip total) to travel to E/M L-1 is too large a challenge on a reusable SSTO when there are so many unknowns about lunar water, propellant conversion, and the means to transport either water or propellants to the depot. Based upon the available information on past programs, the existing capabilities in lander design and reasonable estimations of mass fractions would essentially prohibit the development of a reusable SSTO Human Lander and depot reusable fueling element if the transfer point were E/M L-1 instead of LLO. While it is true that orbit mechanics may limit rendezvous with an LLO depot to only certain times per week or month, the handoff location trade is still dominated by the challenge of a reusable SSTO element capability compared to current technology. There are promising technologies and concepts that result in much more favorable mass fractions, but they are still too immature to carry as the primary architectural concepts at this point.

As mentioned above and to keep focus on affordability, there is a need to develop a reusable architectural element to transport fluids to the LLO Depot. We have chosen a Reusable Water Tank Lander (RWTL) element to transport water to the depot, allowing the orbiting depot to break down the water via electrolysis into H_2 and O_2 for use as propellant (similar to our previous architecture). This architectural decision was made to minimize the storage and transport of two separate fluids, one of which (LH_2) requires a very large containment volume and corresponding structural support on any reusable transportation lunar lander, which is already complicated enough to design as a reusable SSTO element. Electrolysis is a known technology even in space, and while it requires significant power, the mass and volume are small, and it is almost 100% efficient. As well, current solar array densities are such that sufficient high-power arrays are achievable early in the architecture timeframe.

Another element in the architecture, the Cislunar Crew Stage (CCS), acts as a human ferry to move humans from LEO to LLO and back. The CCS will dock with both the LLO Depot and the LEO Depot to transfer crew; the CCS is also reusable with in-situ maintenance and repair. This element's design challenge does not include the need to land on either planetary body, but is used simply for orbit-to-orbit transfer in a relatively benign structural load environment. However, because we have chosen to rendezvous at LEO rather than going directly to the Earth's surface after crew transfer from the LLO Depot, we pay a significant penalty in DV on the return leg back to the Earth by stopping at LEO and not going all the way to deorbit (deorbit would have shed most energy non-propulsively).

This decision was made to rendezvous in LEO for crew exchange to take advantage of the cost, flexibility, and redundant crew launch backup capability to utilize the Commercial Capability Crew concept (abbreviated in this report as CCCrew) as part of the crew rotation transportation concept at the LEO handoff point. *This is the first major change that we have incorporated in our architecture*. Much like CCCrew is being planned for ISS, the Outpost "Users" would buy seats on CCCrew element using the same capability that is being developed now for ISS. The benefit to this architecture is that the cost of a crew seat is competed commercially with existing capability, with use of SLS crew launch if needed in contingencies. The CCCrew element would transfer crew to the LEO Depot, and then the crew would transfer into a docked CCS in LEO. The only change necessary for CCCrew would be to extend the CCCrew element orbit dwell lifetime by docking the uncrewed CCCrew element to a LEO Depot for several months while the crew is at the lunar outpost; power (and atmospheric gases and cryogens if needed) will be provided by the depot. (The LEO Depot provides the fuel for the LEO-to-LLO leg of the CCS journey as well as an interim habitat for the crew while waiting for orbit alignments.) Nominal dwell times would correspond to crew rotations, planned to be every 6 months. This should be easily feasible, even at the cost of additional mass on the CCCrew element, since the LEO orbit requirements for this lunar architecture are less demanding than the ISS orbit (roughly the same altitude but lower inclination for the LEO Depot), meaning that a launch vehicle could launch a heavier CCCrew element to the LEO Depot than it can for the ISS). This operational scenario is described in the Mission Sequence Section of this paper.

The penalty of buying crew seats to LEO as mentioned is the cost of extra orbital energy required for the trip from LLO to LEO for the CCS. It takes roughly a little less than 4100 m/s (including orbit insertion, proximity operations, rendezvous, docking, and margin) to go from LLO to LEO. If it were desired to go directly to the Earth's surface, a small fraction of that DV would be required as a propulsive maneuver because most DV would be dissipated by atmospheric reentry. However, since the technology exists for a reusable CCS to skim the atmosphere to shed energy (shuttle tiles were very effective, although somewhat fragile), we propose to shed almost 2000 m/s using a 2-pass maneuver to travel from LLO to LEO and still have sufficient propellant to insert into the LEO Depot orbit and rendezvous with the LEO Depot. This aerothermal shedding of orbital energy helps significantly because it greatly reduces the LLO propellant demand for the return trip, simplifying the architecture by removing the need to send the additional lunar water to the LLO Depot. In addition, if we need less water at the depot to convert to propellant for the return trip, we need less propellant at the outpost to lift that water to the depot, providing a multiplier savings factor for reducing the orbital propellant need at the LLO Depot. The atmospheric skimming maneuver will be explained more in the Cislunar and Lunar Orbital Elements section. This added function on the CCS is challenging in trying to maintain cold cryotanks while at the same time shedding energy by heating up one part of the element, but we feel confident that the cryotanks on the element can be thermally isolated (especially LH_2, which has a temperature of about 20 K) sufficiently to successfully accomplish this task. Since this is a pure orbital stage, the element shape can be uniquely tailored to its application, which is to ferry crew to LEO from LLO and back, and to use aerobraking on the Earth return leg to reduce the amount of Earth return propellant.

For crew rotation, we buy transport to LEO from a commercial provider, rendezvous and dock with the LEO Depot, transfer crew into the LEO Depot and then to the docked CCS at the proper time. Then the CCS transitions, to LLO and rendezvous and docks with the LLO Depot and transfers crew to the depot. At the right time, the crews swap, and then the HL undocks and lands with the incoming crew on the Moon. For the outgoing crew, they travel in the CCS (which has been fueled at the LLO Depot), undock from the LLO Depot and head back to Earth using aerobraking before rendezvousing with the CCCrew capsule at the LEO Depot. This architecture therefore has no "one-use" crew-rotation element, a requirement for affordability. Looking at the resupply for the two depots, the water resupply of the LLO Depot is performed using a reusable element as well, so there is no "single-use" transportation cost associated with that resupply chain. In fact, moving backward in the lunar water production chain, all of the extraction, transport, processing, liquefaction, and storage hardware is designed to be re-used. Finally, for the Earth leg, we refuel the CCS for the trip to LLO at the LEO Depot. We use commercial launch services to provide water for LEO Depot supply to be converted into propellant (ultimately, we plan to supply the LEO Depot with lunar water). Therefore, the total cost of a crew rotation (not accounting for development and maintenance costs of the re-usable elements) is simply the commercial cost of water cargo to the LEO Depot and the crew seat cost using CCCrew. This first major change to our architecture has allowed us to reduce cost of a crew cycle by roughly half compared to the previous architecture.

One disadvantage of this new crew rotation concept compared to our last architecture is that it requires more lunar propellant, requiring more hardware on the Moon. To offset this need, and to take advantage of the current SLS development program, we shift our emphasis on launch vehicles to use the SLS as the cargo workhorse for the lunar architecture, the *second major change to our architecture*. In taking the SLS payload capability to trans-lunar injection (TLI) orbit (25 metric tonnes (t) for Block 1 and 35 to 40 t for Block 1B), we set the robotic lander size to maximize the payload to the lunar surface without orbital fueling. Depending upon the mission, we use a Block 1 vehicle unless the packaging constraints and/or mass exceed the Block 1 capability to keep costs down since the Block 1 is cheaper than the Block 1B. The choice was made to size the Robotic Heavy Lander (RHL) so that it just fits into the SLS Block 1 TLI capability, and to simplify the operational scenario with a direct launch to the Moon, so that we did not have to rely upon orbital depots and their associated costs early in the buildup phase. We do need depots, but later in the architecture so that we have time to develop them and phase their cost before use. This allows us to double our landed payload mass on the Moon to 5 t per launch compared to our previous architecture.

This additional capability is significant, and has allowed us to deploy larger and more of the modular water elements for lunar ore excavating/hauling, water storage, water processing, propellant processing and storage, and increased power generation. This addition has answered the need for more hardware on the Moon to produce more propellant, and this all has been done within the original total cost prediction of our previous architecture. With the additional mass capability, we have also added a few minor elements to enhance robustness and as a result of additional maturation of the lunar outpost needs (like mobile power sources to keep landers powered).

Although we are launch vehicle agnostic in our overall approach toward lunar architecture in general, we looked at an architecture in this paper using SLS to see how its capabilities would benefit lunar return. There is economy of scale for payload element sizes that we take advantage of in using the SLS as a cargo launch vehicle, allowing bigger and more capable water ice ore excavators and haulers, as well as water processing and electrolysis units. Furthermore, it has allowed us to optimize around the capability of the launch vehicle to TLI, which leads to sizing our major lander elements (HL, RHL, RWTL, and Cargo Lander (CL)). This has resulted in the landers being approximately the same size enough (within a factor of 2) to develop a common small, cryogenic (LOX/LH_2) engine that we can use on all 4 landers. This engine could be swapped out to support the reusable landers, with the RHL and CL being a ready source of engines and other spare parts common to the other two reusable lander elements. We use a complement of 4 engines on each of the reusable landers (HL and RWTL) to allow full engine-out capability, a necessity for reusable landers. For the single-use landers, we use 4 engines on the CL and 2 engines on the RHL. Because the landing and propulsion functions of the two reusable SSTO landers is critical to success, we felt it was necessary to spend early resources to develop such an engine tailored to our application with long life that will become the workhorse engine for the architecture. We also use this same engine for the CCS element and is designed to be changed out in the CCS element in LLO. This common engine is discussed later in the Lander Section.

Currently, SLS is developing the Block 1B Enhanced Upper Stage (EUS). However, for most of the launches that we need to support this architecture, the Block 1 SLS with an Interim Cryogenic Propulsion Stage (ICPS) is adequate and less expensive than the more capable EUS. Therefore, we chose to use a Block 1 SLS for most of the launches. Only when we needed the volume of an 8.5 m shroud or extra TLI capability did we “upgrade” to a Block 1B SLS. We assumed that NASA preserves the capability to launch both versions.

VII. Launch vehicles

The SLS Block 1 and Block 1B vehicles are currently in development. Based upon its designed capabilities and the cost so far, we have high confidence that the latest estimates of performance and recurring cost are accurate enough for this study. The major benefit of the large launch vehicle is to use it as a cargo hauler; the key defining capability that we assume is its 25 t TLI performance capability (net SLS payload). As well, since it is most cost-effective to define a stable flight cadence, we plan for 2 launches per year, generally, with no more than 5 launches in 2 years.

Our new packaging implies design decisions that are made that take advantage of the SLS size and volume that are not easily shifted to smaller launch vehicles should there be problems with the SLS. In this case, reliance upon a single launch vehicle for cargo might result in a significant slowdown in outpost buildup should there be a launch failure or issue with SLS, but since it is cargo only, it has no effect on loss of life. If a problem were to occur with SLS, it would simply delay the development of the outpost. A potential launch vehicle backup that might be considered depending upon the payload need and specific performance requirements would be to launch on a smaller vehicle and fuel/refuel the lunar lander element at the LEO depot if it were available. This option is maintained only as a contingency and can be pursued in future study.

For the larger landers, packaging becomes too significant a challenge even with the 5 m fairing size that SLS Block 1 uses. Therefore, for the HL, RWTL, and CL, we made the decision for strictly packaging reasons to use the SLS Block 1B vehicle, which has a larger fairing diameter (8.5 m) and height with which to package the landers. Since there are re-usable, they are roughly the same size and can use the same engines as mentioned previously, simplifying the architecture as well as saving cost and adding maintenance and redundancy robustness.

Because of the architectural need to provide a single, 10-12 t self-contained structure for human habitation derived mostly from Constellation Program formulation and preserved from our original architecture, we developed a Cargo Lander with an 11 t contiguous payload capability. Because we cannot quite provide a direct path to the lunar surface even with a single launch of SLS Block 1B for that element, we developed a mission design strategy that launches on an SLS Block 1B and then the CL fuels for the final descent to the lunar surface at the LLO Depot. This is scenario is also a good first demonstration of the refueling concept without risking crew, and it occurs later in the architecture to allow time to examine lunar water production and demonstrate the capability to upload to the LLO Depot. We use this heavy Cargo Lander only 3 times currently in the timeline to land two Human habitat elements, along with a Logistics Package that includes the Surface Utility Vehicle (SUV), a mobile pressurized crew rover. We did not consider an architecture that was designed with smaller habitat elements, although it would be a worthwhile trade to see if this was possible to exclusively use the RHL for all outpost development and placement.

The Commercial Crew program has at least 2 separate launch providers, which provides our Lunar Architecture redundancy in case of a launch failure or problem with one of the providers. Our previous architecture relied solely on a heavy lift vehicle to provide crew launch. It could be that certain architectural options could also use the SLS for crew rotation if needed, although the currently defined Orion spacecraft is over-designed for a human ferry to LEO and back to be used reliably in routine crew rotations due to the cost of and SLS launch and an Orion spacecraft; as well, Orion currently does not use LOX and LH_2 propellant, defeating the purpose of a ferry vehicle. However, we have capitalized on the availability of Orion for an emergency return capability in this architecture by stationing a modified Orion spacecraft at the LLO Depot with an upgraded lifetime requirement to allow it to loiter with LLO provided power to last for 5 (minimum) to 10 (goal) years.

Summarizing, this architecture can be solely developed by the U.S. and launched using the SLS. While we are ultimately vehicle-agnostic from an overall perspective, for the purpose of this analysis, we have taken advantage of 2 capabilities that appear to be on the path toward first launch in a few years. The SLS capability allows architectural decisions that take advantage of economy of scale for the lunar landers and the payloads that are carried by them so that a single launch allows for a larger lunar payload element should that size option provide more efficient performance (lower cost per kilogram). We believe that using the SLS as a large cargo carrier benefits our architecture in many key capabilities.

We also believe that a number of international partners will want to contribute pieces of the required infrastructure. Since their possible contributions are undefined, it is difficult to predict how they would affect the launch vehicle picture, but our architecture is meant to be incremental and cumulative over time. Given specific agreements, it could be envisioned that the U.S. performs all launch services for architecture elements. It could also be envisioned that international partners would want to use their own launch vehicles for their contributions. There are many options that are enabled by the development of a heavy lift vehicle like the SLS, but by no means is it the only launch solution.

VIII. Cislunar and Lunar Orbital Elements

In this implementation of our architecture, unlike our previous one[2,3], we employ a reusable Cislunar Crew Vehicle (CCS) to ferry the crew and supplies, crew-unique EVA hardware, etc from low Earth orbit (the LEO Depot) to low lunar orbit (the LLO Depot) and back. Previously, we relied upon the Crew Exploration Vehicle (now Orion) for the return leg to the Earth, coupled with a throwaway TLI stage on the way to the Moon. The new implementation minimizes the recurring cost of a crew rotation, which is expected to be the more significant cost as the lifetime of the architecture is large compared to the development time (i.e., several dozen to over 100 crew rotation cycles during the lifetime of this program). It is expected that this lunar architecture is sufficiently enabling to continue operation continuously (much like the Antarctica model but with commercial interests as well) as both International Partners and commercial entities ramp up while NASA shifts focus to new challenges after it has completed assessment of each of the objectives mentioned above in section VI.

The primary function of the CCS is to transport crew from LEO to LLO and back, with crew ingress/egress at the two end point depots. Further, to minimize propellant mass, the CCS is sized with a full propellant load for only a single leg at a time. In addition, to make most efficient use of mass, we assume a fully integrated propulsion and attitude control system, fueled by LOX and LH_2 because of the objective of using lunar propellant for the return leg. There are technology plans in place for such a coupling of these two systems, and should not be as challenging an issue as others in this architecture. As well, LOX and LH_2 propulsion systems are most efficient for orbit changes outside of the atmosphere, with high ISP and thrust that can only be achieved using LOX/LH_2 systems. Since it is reusable, and to minimize need for reliance from Earth, we use propellant-generated pressurants so that we do not need to use helium or nitrogen, which would need to be replenished every rotation. While we note that it is probable that some amount of other species might also be harvested on the Moon[9], we will view those as bonus products that add flexibility at the outpost; nitrogen in particular can be used for atmospheric revitalization in the habitable volumes. To further reduce the propellant demand for the return leg (return leg propellant is totally provided by lunar water upload to the LLO Depot), we use an atmospheric aerobraking concept to non-propulsively shed orbital energy (DV) in order to maneuver to the LEO Depot with just the right amount of propellant remaining (including sufficient margins).

We use a value of 4066 m/s of DV for TLI, LOI, and orbit adjustment/correction, equivalent attitude control, and margin for the LEO-to-LLO (at 100 km) leg, the more propellant-demanding leg; we add 350m/s for orbit phasing using interim elliptical orbits to arrive in the proper orbit at the right time. On the Earth return leg, we shed energy with aerobraking, allowing a reduced total propulsive DV requirement of 2250 m/s for the return leg from LLO to LEO, also adding 350 m/s for orbit phasing and alignment. We believe that with a two-skip atmospheric maneuver, at least 1,800 m/s (and possibly as much as 2000 m/s, to be studied further) can be shed to allow successful rendezvous and dock at the LEO Depot. Due to atmospheric density uncertainty at the time of the maneuver, the first orbit pass skims the atmosphere to reduce the major part of the DV with a wide band on the velocity change while collecting atmospheric density data to be used in the calculation of the second pass. The second pass would be a more precise pass to trim just the expected amount of energy from the CCS. We believe that aerobraking should allow up to 2000 m/s to be shed successfully and accurately enough for this use; we assume a design requirement of 1900 m/s (plus or minus a narrow band reserve for the two-skip maneuver to preserve design and operational margin). Not shedding enough energy would result in needing to use more propellant to shed energy to get to the LEO Depot orbit and it would perturb the rendezvous phasing as well. Reducing the DV too much requires additional propellant to boost the orbit high enough (add orbital energy) to rendezvous with the LEO Depot.

We size the dry mass to be as small as practical, so that the propellant demand for return lunar propellant is minimized. The more structure, the more propellant, and it is a highly non-linear function governed by the rocket equation. As a starting point for sizing, we use the Apollo CS/SM combination dry mass[10] of roughly 12 t for 3 crew and we allocate and up-size a reusable CCS at 17.5 t for 4 people, with a single structure (no need to separate and land a CM capsule). The additional mass (compared to Apollo) can be used to add a 4th person to the crew complement, increase the crew transfer mass from 1,050 kg to 2,000 kg (total transferred), build efficient structure to shed energy during aero-passes while minimizing the thermal coupling to the propellant tanks, and add cryo-cooling for both LH_2 and LOX to minimize boil-off loss. With existing plans and technology, we feel that a 20 K cryocooler is achievable so that a small percentage (less than a few %) of LH_2 propellant would be lost during the cislunar transfer. We calculate that roughly 30 t of usable propellant (plus 3.6 t for 350 m/s orbit phasing propellant) is needed for the Earth-to-LLO segment using the rocket equation:

$$M_f = M_0 * \exp(-DV/[9.8067 * Isp]), \text{ where:}$$

a. M_f = Final Mass
b. M_0 = Initial Mass
c. DV = delta-velocity
d. Isp = Isp of the propulsion system

On the LLO-LEO segment, since we shed roughly 1900 m/s using aerobraking, we only need about 16 t of usable propellant from the Moon.

The CCS would be stationed at the LEO Depot when not in use, and would rely upon depot power to charge batteries and maintain long term storage for propellants in the CCS (i.e., power the cryocooler). The CCS would be designed to be maintained at either depot, for both routine maintenance/inspection as well as repair and replacement. The first use of the CCS would validate the aerobraking strategy while the CCS is not crewed, providing additional confidence that the CCS can perform its intended function (see MISSION SEQUENCE section for details).

For this element, we also assume a common engine with some of the landers, which allows flexibility for maintenance and repair, reduces total part count, and reduces the cost of engine procurement. We preserve the capability of single engine out for the entire journey for both legs, enhancing crew survivability. The only change that might be entertained between this engine and the lander engines might be larger nozzles (corresponding to slightly higher Isp), because of the packaging freedom with the CCS as opposed to the landers; these elements would remain in orbit for their entire lifetime and can afford the packaging space to have large nozzles. CCS requirements and characteristics are summarized in Table 2.

Table 2. Cislunar Crew Stage Characteristics

Cislunar Crew Stage (CCS)	
Characteristics	**Rationale**
Purpose	To Transport Crew and a small amount of crew supplies from LEO Depot to LLO Depot and back with refueling for each leg separately. This element would be refueled at both the LEO Depot and the LLO Depot. In performing this function, the CCS would need to aero-brake in the Earth's atmosphere to remove a large part of the DV energy required to return to the LEO Depot.

Table 2. Cislunar Crew Stage Characteristics (continued)

Cislunar Crew Stage (CCS)	
Characteristics	**Rationale**
Mass:	As a starting point, use the Apollo CM+SM combination, which was roughly 12 t. Subtract mass to remove capsule structure to touch down in the Ocean, but add back for cryo propellants and storage, reusability, arrays, and heat shield/thermal design to keep heat off of LH_2 tanks (and O_2 tanks).
- Dry Mass	17.5 t
- Crew Mass	2 t - for 4 crew plus EVA unique crew-fit needs and miscellaneous)
- Prop Mass	33.6 t - LEO-to-LLO leg, Usable (350 m/s DV for orbit plane change/phasing)
	16 t - LLO-to-LEO leg, Usable (350 m/s DV for orbit plane change/phasing)
Operations Concept	Addressed in the Mission Sequences Section.
Notes	Requires a pressurized docking port (maybe two for redundancy). Requires essentially zero boiloff cryo cooler technology for both LOX and LH_2, and cryo RCS. Must use propellants as pressurants. Must refuel at LEO and LLO Depots. Requires a heat shield to absorb aerobraking heat, but must be thermally decoupled from the cryo tanks (at the other end). Uses the common engine, in a 2 or 4 cluster, with single engine out capability. Must be able to change out engine in orbit. Remains in orbit for its entire lifetime (does not need to land or splashdown). Leverage Constellation and post-Constellation Orion work on Crew Systems, Avionics, and Life Support to lower development cost.

Depots would be located at both transportation end points in cislunar space; the first is at the LEO location, and the second one is in LLO. To keep costs down, both the LEO and LLO Depots use essentially the same or very similar design; the only challenge might be to tailor the thermal design based upon unique thermal environments associated with both locations. A single depot design eases operational and maintenance burdens, lowers unique part counts, simplifies interfaces, and reduces cost so that the second unit is simply the unit cost, with no (or very little for thermal customization) Non-Recurring Engineering (NRE). With the use of SLS as the lunar surface architecture workhorse, there is no longer a need for the LEO Depot to fuel any elements in the buildup of the lunar surface architecture, another plus for this architecture. In the previous architecture, the LEO Depot was required to fuel roughly 8 t for every RHL lander during the outpost buildup. As well, for the large Cargo Landers, the previous LEO Depot provided 60 t of usable propellant for the Cargo Lander missions, driving the size of the LEO Depot to be larger than the LLO Depot. In this case, the LEO Depot is only used for crew rotations. The LEO depot provides 33.6 t of usable propellant to the CCS per crew rotation leg. One additional function that this architecture adds to both Depots is loiter of crew for a few days within a pressurized volume at the depot. We made this addition because of orbit plane alignment challenges with a choreographed crew handoff, so that the CCS now simply bounces from depot to depot roughly every 6 months with crew handoffs. This additional depot capability essentially adds short periods (usually a few days) of crew presence in LEO as well as in LLO. While we do not take advantage of this capability other than routine or unscheduled EVA maintenance and repair activities at each depot location, it enables many cislunar activities not directly tied to this architecture. The LEO Depot would be stocked by the new CCCrew capsules that rendezvous and dock to the depot every crew rotation, and some of the supplies will be carried forward to the LLO Depot in the CCS.

The LLO Depot is a multi-use depot in this architecture. It first provides propellant for the CL, which is presently used 3 times for the largest human habitation payloads landing on the Moon. (Nothing precludes more cargo landers for future lunar surface missions, depending on how the outpost evolves.) The CL mission scenario launches the CL plus payload with enough fuel to arrive at the LLO Depot to fuel for the CL lunar landing (11 t of usable propellant). This CL fueling also provides an end-to-end demonstration without crew of the entire concept of using lunar water to fuel one of the architecture elements. The demonstration will show that lunar ice ore can be harvested, that water can be extracted, stored, and uploaded to the LLO Depot, electrolyzed and liquefied into LOX and LH_2 cryogens, and stored effectively with essentially no boil-off loss for months at a time. The demonstration will also show that fuel can be efficiently transferred to a lunar architecture element (CL) with minimal loss (<1%), and that the element can use it successfully in performing its propulsion functions.

The second use of the LLO Depot is to re-fuel the CCS for the return journey with crew to LEO to rendezvous with the LEO Depot. In this case, the HL is already docked to the LLO Depot with crew inside the Depot. The CCS is docked to the LLO Depot, refuels (16 t of usable propellant), and when orbit phasing is proper, the crew ingresses into the CCS, the CCS undocks and performs the first TEI maneuver to line up the orbit plane for its way to Earth.

The third function for the LLO Depot would be house the emergency Orion return vehicle, which would be docked in a long-term condition at the LLO. The Orion purpose would be to act as an emergency lifeboat to bring a crew back to the Earth's surface, bypassing the LEO rendezvous. In this scenario, only power and exchange of pressurized air volume is required. Because the Orion capsule sheds energy via atmospheric reentry, it does not need to worry about LEO orbit phasing, since it would directly enter Earth's atmosphere and land. In this scenario, the Orion lifeboat uses storable propellants that have long lifetimes.

Finally, the depots provide habitable volumes for crew loiter in a pressurized environment for short periods of time (a few days) for optimum orbit phasing, but could provide extended emergency or unscheduled maintenance stays for longer. This is an added function, and greatly enhances the usefulness of the entire architecture. It allows repairs on the CCS, a critical reusable element in the architecture and therefore would have a few Extra Vehicular Activity (EVA) suits for external maintenance.

In order to validate that our concept that a depot can be reasonably designed and packaged for a single SLS-1B launch, we discuss the overall layout and packaging for the SLS configuration. To size the cryogenic tanks, we choose the CCS Earth return design case so that the tanks are large enough to hold the fluid for the CCS refueling scenario needing 16 t of usable propellant. Because the fuel Mixture Ratio for these cryogenic engines is lower than the electrolysis ratio of 8:1, there will necessarily be more oxygen produced than is needed for refueling of any of the architecture elements. Typical Mixture Ratios for the RL-10 range from 5.5 to 5.9, which means that there will be excess LOX produced in electrolysis compared to H_2. The more critical fluid to plan for is LH_2. Using an MR of 5.5, roughly 2.45 t of usable LH_2 is required for the CCS Earth return leg, with roughly 13.6 t of LOX. To add for losses, Performance Reserve, and Margin, assume that 2.6 t of LH_2 is required for Earth return CCS refueling (along with 14.3 t of LOX). To size the tank, we choose a volume sufficient to store 4 t of LH_2, which should provide sufficient margin (50%) at this conceptual level. Using LH_2 density of 70.8 kg/m^3, and a double-walled LH_2 tank to minimize heat coupling, and to package around a 8.5 m diameter SLS Block 1B vehicle, that would result in a tank height of roughly a few meters at most, if the tank internal diameter was 7 m even with a double-wall design if that design were used. This double wall LH_2 isolation can be addressed as a trade study in the future, but with long life reusability needs, the added insulation of a double walled tank should substantially reduce the cooling cost to maintain zero boil-off (mass versus cooling power trade). For LOX with a density of 1140 kg/m^3, we choose a volume sufficient to store 30 t, with a diameter of 8 m (don't need a double wall for LOX). In this case, the height for the LOX tank is less than a meter, should the design choose a diameter the size of the SLS vehicle. Given these volumes for the two tanks, there is a lot of packaging flexibility for the tanks within the depot. For water, the Robotic Water Tank Lander would upload 8 t of water at a time to the LLO Depot, so it should be at least large enough to hold 8 t plus margin (assume a density of 1 t/m^3). We choose a volume sufficient to store 17 t of water, enough for two loads of water from the Moon. Since the water tanks would be unloaded during the launch, and since there is no need to thermally cool the tanks, they can be contoured to best fit the packaging scheme chosen for the depots.

For size of the solar arrays, we use the conversion factor of 5 kW/kg product (C. Mittelsteadt, personal communication, 2016) to convert water to O_2 and H_2. This electrolysis process should be essentially no loss, with the assumption that imperfect reaction gases can be looped back into the input stream for a continuous operation. With 150 kW dedicated to electrolysis, this would provide roughly a little more than 14 t/month of electrolyzed product (assume Earth or Moon occultation at 35 %). To add mission flexibility, we choose a 200 kW array to allow for other loads like cryocooling and docked element power. While this seems like a challenge, current and near future arrays will be able to achieve these capabilities with efficient packing densities (T. Kerslake, personal communication, 2016). For packaging into the SLS launch vehicle, an SLS Block 1B fairing offers significant flexibility in packaging as a result of the relatively small size of the tanks and the possibility of starting the fairing above the tanks if more volume is needed to package the solar arrays, habitable volume, docking ports, batteries/fuel cells, etc). Note that orbit maintenance would be required at both LLO and LEO depots, and the LLO Depot would also need to perform all orbit maneuvers to achieve the final LLO orbit, so that propulsive engines are required. Future study would size and locate the engines and determine whether or not the attitude control function can also be performed with the same system. Note that orbit maintenance on the LLO Depot is roughly 500 m/s of DV per year. Also for future study, there is surplus LOX due to the Mixture Ratio being different from the electrolysis ratio. This excess fluid might be usable to maintain the orbit of the Depot "for free", with the recognition that the Isp for cold gas O_2 propulsion is low and oxygen is very reactive chemically in systems, which is why it is not selected as a monopropellant in flight applications. However, for this case, it would be a free fuel that would probably need to be vented periodically anyway.

Since the CCS will be parked at the LEO Depot for long periods of time, we choose to use some of the propellant storage capacity in the CCS itself to store its propellant and therefore we do not need to size the Depot tanks to hold all of the propellant for CCS from LEO-to-LLO scenario. That implies that the re-fueling at the LEO Depot will necessarily require dwell time for some water to be electrolyzed and stored in the CCS tanks as it is produced at the LEO Depot. This limitation does not affect mission performance, since in all cases, there is not an immediate unplanned need for a fully fueled CCS when it is docked at the LEO Depot. Any unplanned immediate needs for a crew rescue or unplanned Earth return are handled by the Orion capsule, docked at the LLO Depot. Immediate crew needs in LEO can be handled by the CCCrew capsule for an emergency return to Earth. Finally, future study can examine the benefit versus drawback of having larger tanks in the depots at the expense of higher cryocooler energy needs, assuming packaging can be met to fit into the SLS. Depot requirements and characteristics are summarized in Table 3.

Table 3. Depot Characteristics

Depot (LEO and LLO)	
Characteristics	**Rationale**
Purpose	To act as an interim crew station and refueling depot for LO2 and LH_2 in 100 km Low Lunar Orbit (LLO) and in 400 km LEO (same overall design) for any element connected to it. Depots will have pressurized volume for up to 8 crew for short periods of time (<7 days). Depots will also produce and store propellants with minimal loss (<1 – 2% per several months) from water uploaded to the Depots. Depots will also provide Integrated Depot Assembly (Depot plus attached elements) attitude control; Depots will provide power to Elements when docked to the Depot. Depots will provide Orbit stationkeeping (LLO orbit annual DV is 500 m/s).
Mass:	Start with TLI Mass (35 - 40 t), and assuming EUS performs TLI, calculate propellant for LLO Propellant Depot to perform LOI and LLO burns to get to final orbit. For LLO Depot, 8 t for propellant (with 1 t margin), leaving 29 t for dry mass (habitat, power, communications, etc).
- Dry Mass	29 t
- Crew Mass	N/A
- Prop Storage	30 t LOX (~27 m^3)
	4 t LH_2 (~57 m^3)
	10 t Water (~10 m^3)
Power	Looking at available state of the art, should be able to provide a deployable array system for 200 kW, packaged using an array packing parameter of 40 kW/m^3. Earth Occultation is assumed to be 35% of the orbit. Power will only be for housekeeping during eclipse (no electrolysis).
- Electrolysis	150 kW
- Cryo Cooling	25 kW
- Miscellaneous	10 kW
- Attached Elements	15 kW
Characteristics	**Rationale**
Operations Concept	For the LLO Depot, the SLS Block 1B with EUS launches an empty Depot except for LOI and LLO fuel (roughly 8 t) to TLI (37 t total). From there, EUS falls off, and the LLO Depot performs the LOI burn and the LLO burn to get into initial orbit position. Solar Arrays unfurl and ops begin. For the LEO Depot, the SLS Block 1 (with no upper stage) launches a full Depot to LEO orbit, which is then trimmed and circularized by the LEO Depot. The Depot would then deploy arrays and begin operation when the Commercial Cargo vehicle is ready to send up water.

Table 3. Depot Characteristics

Depot (LEO and LLO)	
Characteristics	**Rationale**
Notes	Requires at least 4 pressurized docking ports (small diameter), and an airlock. Ports can interface with either a pressurized interface or non-pressurized interface to transfer fluids. Requires essentially zero boiloff cryo cooler technology for both LOX and LH_2, and cryo RCS. Must tap into cryo tanks to provide prop for LOI and LLO burns, and stationkeeping for both Depots. Explore use of double wall LH_2 tank. Must provide redundant cryo fueling capability. May require radiator. Remains in orbit for its entire lifetime (does not need to land or splashdown). Must provide communications with Earth and Outpost. Must allow engine change out of a docked Element engine. Must accommodate different inertias for stationkeeping and attitude control. Must allow docking of Elements to Depot, with crew ingress/egress. Must have deployable arrays. Contains crew pressurized volume for 4-8 crew for short periods. Because of excess supply of LOX, investigate use of LOX thrusters for station-keeping. All standard EVA hardware can be stored in the Depot but EVA unique crew-fit needs will be orchestrated with crew cycles.

IX. Lunar Landers

As mentioned in section VI, the choice of transportation node point was made based upon the need to minimize the design challenges associated with the two reusable landers and their missions, the HL and RWTL. Because both are reusable, SSTO cryogenic elements that are fueled with propellant from the Moon, have the most challenging requirements. There has been public discussion suggesting that the best nodal rendezvous point would be the Earth-Moon (E-M) L-1 position for ease of operational flexibility. This is true in that the E-M L-1 point is operationally easier to use than LLO; however, when using this architecture with reusable, cryogenic, SSTO landers, it becomes impractical to meet the additional DV requirements between LLO and E-M L-1 with a single stage, using what we feel are reasonable mass fractions that can be achieved currently with known design capability. The particular mass fraction we note for this study is the ratio of dry mass (not including payload) to propellant mass, and this ratio we call the Dry Mass Fraction (DMF; it is sometimes called the scaling equation coefficient). This DMF can range widely, and varies for different propellants. While there have not been landers that have been built and flown using LOX/LH_2, there have been a few studies by NASA that have matured to preliminary design, most notably the Altair lander from the Constellation Program[11]. The Altair design went through a long, protracted study and maturation phase, and therefore was further than a Preliminary Design maturity for the lunar mission prior to closure. The Dry Mass Fraction for the Cargo lander variant was roughly slightly under 0.5 (0.47), which is quite efficient for cryogenic landers. However, this lander never reached full maturity and was designed without the necessary engine-out and reusability needs that this lander will require. Based upon past engineering, it is wise to assume that a reusable SSTO cryogenic lander will have a DMF slightly above 0.5.

Note that one penalty for the high Isp of LOX/LH_2 is this high mass fraction. For the descent stage of the Apollo Lunar Module, using a different (low Isp) propellant, the DMF was roughly 0.26. However, we gain substantially as shown in the rocket equation for mass efficiency during DV changes using the higher Isp for LOX/LH2 as well as the ability of fueling with lunar propellants, so there is no additional trade that we can perform if we want to use lunar propellants. This is also quantitatively addressed later.

One additional conceptual design was performed for the second mission of the Robotic Lunar Exploration Program (RLEP), which included a cryogenic lander for cargo, although smaller than the Altair lander. To give a lower limit to the ratio, the RLEP-2 lander DMF was roughly 0.5 for a non-human rated, single use cryogenic lander with a total Mass to TLI of 9600 kg. This concept probably reached a Preliminary Design maturity level, and so there was some dry mass growth due to design immaturity that would still be realized.

The rocket equation is highly non-linear and very sensitive to this DMF parameter. Choosing a DMF of 0.50, for a 30 t RWTL as a hypothetical exercise, we can send roughly 5.6 t of water to LLO and return empty, with a propellant mass of 16 t and an empty water tank (500 kg). If we increase the DMF to 0.6, for the same payload to LLO, we need a mass of roughly 75 t, with a propellant load of roughly 43 t! Based upon this very sensitive parameter, we have developed a suite of landers best optimizing the DMF with the lander functions and requirements.

In addition to this more in-depth discussion of DMF, we have adjusted all of the lander sizes to take advantage of the larger launch vehicle, the SLS. For the Outpost buildup, this manifests as a larger, more capable Robotic Heavy Lander (RHL), with essentially double the landed payload capability (5 t) compared to the previous architecture (2.5 t). This is achieved by using the advertised TLI capability of the SLS Block 1 vehicle, and choosing a lander that maximizes the landed payload mass while just fitting into the TLI envelope. This single-use lander is the workhorse of the outpost buildup and its operational concept has also been changed to remove the need for fueling at the LEO Depot on the way to the Moon. This single-use lander design is used 15 times in our architecture (further maturation will need to figure out the outpost logistics to land each one of the 15 landers). Additionally, the use of SLS offers the potential to allow outside payloads to use available space (mass) on the landers depending upon the mission, providing NASA with a bargaining chip to use to further the benefit of the overall architecture. For instance, extra space could be bartered with an international partner or with a water provider to the LEO Depot, or even for compensation value for crew rotations within the CCCrew program. Our selections of size, Isp, DMF, and Payload for each of the landers is provided in Table 4.

Table 4. Lander Characteristics

Landers	
Characteristics	**Rationale**
	ROBOTIC MEDIUM LANDER (RML)
Purpose	To deliver the RTG-powered Water Ice Explorer (WIE) to the surface at the lunar poles (one lander per WIE per pole). Assuming that the WIEs are powered by RTGs, there is no keep-alive function for the landers. If that is not the case, the landers will need to add power generation and WIE charging capability.
Mass:	In the initial sizing done for RLEP-2, these landers were derived from Architecture 3 in that the landers were cradles, with a small solid to take most of the DeltaV away and then fall off. The Dry Mass is roughly 1200 kg/lander, with a Payload Capacity of 500 kg.
- Dry Mass	1.2 t
- Crew Mass	N/A
- Prop Mass	1.88 t
- Payload Mass	0.5 t
- Dry Mass Fraction	N/A; not cryo
ISP:	Non-cryo fuel
Operations Concept	For both landers, launch on a single SLS launch to TLI Orbit (along with Communications cluster). At the right phasing, deploy one and then the other lander to the correct pole. Also detach and disperse the communication satellite cluster to provide almost total coverage of both poles.
Notes	This lander is not a cryogenic fueled lander. If the WIE cannot use an RTG, then they must be battery operated, and this power generation function would need to fit on the lander. That would grow the landers. However, there is still much margin on the SLS Block 1 TLI performance for this mission, so the lander can grow and still have plenty of capability left over (currently, total TLI mass, 10 t, with an SLS TLI performance of 25 t, so there is ample reserve mass. If an RTG is used, the WIE batteries might be sufficient to handle the RML needs in the transit leg prior to landing; this should be assessed.
	ROBOTIC HEAVY LANDER (RHL)
Purpose	To deliver large payload (5 t) to the lunar surface (not reusable) by performing every propulsive maneuver downstream of TLI. Payloads can either be fixed or removable via payload locomotion. RHL will by a LOX/LH_2 lander, and will use gimbaled Common Engine(s) that might be scavenged after mission completion. RHL will use integrated Main Propulsion and Reaction Control System (same LOX/LH_2 for attitude control).
Mass:	Based upon an RLEP-2 Lander Architecture trade, these landers were derived from Architecture 9 in that the landers were bigger and used LH_2 and LOX as propellants. Dry Mass Fraction is 0.5 (Dry Mass/Prop Mass).
- Dry Mass	6.3 t
- Crew Mass	N/A
- Prop Mass	13.7 t
- Payload Mass	5 t
- Dry Mass Fraction	0.5

Table 4. Lander Characteristics

Landers	
Characteristics	**Rationale**
ISP:	440 (Main), 300 (RCS)
Operations Concept	The SLS Block 1 with iCPS launches a fully loaded RHL (25 t). The iCPS does the TLI and then the iCPS disposes. The RHL does the LOI, then the LLO circularization (if needed), and then descent to the surface. Once on the surface, any removable payload will drive off of the lander under its own power.
Notes	Payload capability assumes that some additional lifetime for the lander is needed after payload deployment to keep engine alive for future use; Solar arrays, heaters, and batteries would count against the 5 t payload. Investigate a cradle design instead of landing legs, similar to the cradle landers in the RLEP architecture study which had a low payload CG for ease of removal. Part of the fixed payload will be solar arrays that want to get as high as possible, so RHL will need to accommodate vertical arrays.
	ROBOTIC WATER TANK LANDER (RWTL)
Purpose	To deliver water as payload produced on the lunar surface to the LLO Depot in 100 km LLO and then return to the surface for refueling and reuse. Capable of refueling at LLO Depot in contingency. Water transfer will be pumped from water tanks on RWTL. RWTL will be a LOX/LH_2 lander, and will use gimbaled Common Engine(s) that might be Removed and Replaced by crew at the Outpost. RWTL will use integrated Main Propulsion and Reaction Control System (same LOX/LH_2 for attitude control).
Mass:	Use Rocket Equation and efficient structure (0.5 Dry Mass Fraction), with Isp of 445 (Main) and 350 (RCS). Water tank can be contoured to best reduce Dry Mass Fraction, and no payload removal from the RWTL except liquid water transfer.
- Dry Mass	11.9 t
- Crew Mass	N/A
- Prop Mass	23.6 t
- Payload Mass	8.5 t (8 t water, 0.5 t tankage and baffles)
- Dry Mass Fraction	0.5
ISP:	445 (Main), 350 (RCS)
	ROBOTIC WATER TANK LANDER (RWTL) - Continued
Operations Concept	The SLS Block 1B with EUS (need Block 1B for the 8.5 m fairing volume) launches a fully prop-loaded RWTL (25 t), but no water as payload. The EUS does the TLI and then the EUS disposes. The RWTL does the LOI, then the LLO circularization, and then descent to the surface. In this case, the RWTL is has no water except for the 500 kg of dry water tankage. See notes on TLI. Once on the Moon, RWTL will interface with and receive power from a mobile Lander Support Cart (LSC) when on the lunar surface to keep healthy. LSCs will contain the power generation function so that the landers are streamlined for just its intended propulsive functions plus small power generation for in-situ power. When getting close, the RWTL gets fueled fully with Prop for a round trip from the Fuel Tanker, as well as water from the Water Tanker to send to the LLO Depot.
Notes	Requires essentially zero boiloff cryo cooler technology for both LOX and LH_2, and cryo RCS. Storage power while on ground at Outpost provided by LCS. Berms will prevent direct solar impingement on cryo tankage. Must use propellants as pressurants. Uses the common engine, in a 2 or 4 cluster, with single engine out capability. Must be able to change out engine at Outpost.
	HUMAN LANDER (HL)

Table 4. Lander Characteristics (continued)

Landers	
Characteristics	**Rationale**
Purpose	To ferry human crew from LLO Depot to the lunar surface and then back up from the lunar surface to LLO Depot when the human crew is ready to return; the HL is reusable. Capable of pressurized docking (and refueling in an emergency) and crew ingress/egress at LLO Depot. HL will be a LOX/LH_2 lander, and will use gimbaled Common Engine(s) that might be Removed and Replaced by crew at the Outpost. HL will use integrated Main Propulsion and Reaction Control System (same LOX/LH_2 for attitude control).
Mass:	Use Rocket Equation (0.56 Dry Mass Fraction), with Isp of 445 (Main) and 350 (RCS). Crew lander is more constrained, and therefore we use a more conservative Dry Mass Fraction. Crew services will be minimal; dwell time is short.
- Dry Mass	10.74 t
- Crew Mass	2 t down to Outpost; 1.5 t up from Outpost
- Prop Mass	19.8 t
- Payload Mass	N/A
- Dry Mass Fraction	0.56
ISP:	445 (Main), 350 (RCS)
Operations Concept	The SLS Block 1B with EUS (need Block 1B for the 8.5m fairing volume) launches with a fully fueled (but uncrewed) HL (31 t). The EUS does the TLI and then the EUS disposes. The HL does the LOI, then the LLO circularization, and then rendezvous and docks with the LLO Depot. It waits for the crewed CCS to dock to the LLO Depot. Once the CCS is ready with a crew complement, the HL receives crew from the CCS, docked at the LLO Depot. Once the crew has ingressed, the HL will undock and descend to the surface. After landing, the Surface Utility Vehicle (SUV) drives to the HL and docks to it, allowing a shirt sleeve egress path from the HL to the SUV. At the Outpost, the HL will interface with a mobile LSC when on the lunar surface to keep healthy and preserve any unused propellants (power provided by LSC). Lander Support Cart will have deployable arrays so that the landers are streamlined for just their intended functions.
Notes	Requires essentially zero boiloff cryo cooler technology for both LOX and LH_2, and cryo RCS. Storage power while on ground at Outpost provided by LCS. Berms will prevent direct solar impingement on cryo tankage. Must use propellants as pressurants. Uses the common engine, in a 2 or 4 cluster, with single engine out capability. Must be able to change out engine at Outpost. Because it is reusable and with a need to streamline lander mass, only the critical human functions need to be provided by the HL; small power generation capability during primary mission activities. Emergency response will require pressure suits and suited egress, but normal operations will be shirt sleeve. Crew Mass down includes EVA unique crew-fit needs
	CARGO LANDER (CL)
Purpose	To deliver 11 t of large payload to the lunar surface. Assume that the engines can be removed for future use. Design should allow for large structures to be permanent, whereas smaller payloads can be removed. Must dock and refuel at LLO Depot. CL will by a LOX/LH_2 lander, and will use gimbaled Common Engine(s) that might be Removed and Replaced by crew at the Outpost. CL will use integrated Main Propulsion and Reaction Control System (same LOX/LH_2 for attitude control).
Mass:	Break analysis into two segments: One segment getting to LLO Depot. One segment getting from LLO Depot down to Outpost. Start with scaling up Dry Mass from RHL (11 t payload/5 t payload) X RHL Dry Mass = 14 t. That sets Dry Mass. Then launch with full tanks on SLS Block 1B and replenish at LLO Depot. Single use only.
- Dry Mass	14.0 t
- Crew Mass	N/A
- Prop Mass	16 t (5 t to perform LOI and LLO, then refuel 11 t to fill tanks at 16 t for landing)
- Payload Mass	11 t
- Dry Mass Fraction	0.8 (this is because we are refueling at LLO Depot)

Table 4. Lander Characteristics (continued)

Landers	
Characteristics	**Rationale**
ISP:	445 (Main), 350 (RCS)
Operations Concept	The SLS Block 1B launches a partially fueled CL (fuel up to EUS TLI capability). The EUS does the TLI and then disposes. The CL does the LOI, then the LLO circularization, and then rendezvous with and docks for refueling to the LLO Depot (11t additional propellant). Once fueled, the CL undocks and descends to surface.
Notes	Requires essentially zero boiloff cryo cooler technology for both LOX and LH_2, and cryo RCS. Must use propellants as pressurants. Uses the common engine, in a 2 or 4 cluster, with single engine out capability. Must be able to remove engines at Outpost. Because of large payload size, need for mobile payload egress, and the need for low Dry Mass Fraction, a horizontal lander with low Cargo CG should be investigated. Mobile payload egress will be performed by the element.

One additional decision that we have made is the selection of a common engine for all of the landers, as well as for the CCS as mentioned earlier. For the two reusable landers, we preserve full engine out capability. For design margin, we assume a net Isp of 450 seconds for this reusable engine. We feel that this should be easily achievable with comparison to the suite of existing RL-10 engines, with Isp ranging from 446 to 465 seconds. Depending upon the need for throttling (and reduced Isp with lower throttling), the actual rated Isp for the new common engine at full thrust would be higher (455 to 460). However, we would trade Isp above the 455 to 460 range for added lifetime, reduced cost, and/or restart margin because of the desire to have high reliability, many restarts, deep throttling, and low cost. To use an analogy, we would want to build a versatile pickup truck as opposed to a Ferrari, which is built to maximize performance. In calculating the engine size, and assuming that we have a strategy of opposing engines being shut down in case of engine failure for a 4 engine configuration for both reusable landers, we use roughly a 1.3 thrust to weight ratio in the calculation. This 10,000 lbf (44.5 kN) – 11,000 lbf (50 kN) engine would be a new development and a single procurement for all engines. This engine is a little less than 50% of the thrust of a RL-10B-2 model. Our intent is to tailor this engine to fit in a smaller space and allow larger engine bells where possible to tickle out the best Isp performance with a reliable engine that can be more easily handled on the lunar surface. It is required that engines will be removed and replaced over time for each of the reusable elements in the architecture, and therefore the smaller and less massive the engine, the easier it is to remove.

One parameter for future study is the orientation of landers, either horizontal or vertical. Horizontal refers to the orientation where the payload is low to the ground, encased in a can-type structure that lands on its side[12], but sits in the SLS vertically. This is as opposed to landing in a more traditional vertical mode, where the lander is oriented in the SLS as it will land on the Moon, with all the engines clustered on the bottom[11]. The benefit of a horizontal lander would be that it is much easier for the payloads to exit the lander. Another benefit might be, depending upon the design, that the engines and landing gear could be better designed to distribute the loads and potentially to gain additional nozzle extension and improve Isp. In fact, existing studies[12,13] by ULA seem to indicate that the DMF can be improved for a horizontal lander that is designed in more of an upper stage configuration. We have not chosen to incorporate those ideas yet, although they look promising and definitely should be studied further to see if they have merit upon further maturation.

X. Lunar surface assets

The most significant change for the lunar surface elements is the growth in the lunar water-to-propellant chain to increase total production. For the Excavator/Hauler (EH), we increased the mass allocation by almost a factor of 2, which is double the estimation of this capability back in the first NASA Lunar Architecture Team exercise (LAT-1). Mining operations probably constitute the largest uncertainty in the entire architecture; how far does the EH need to travel to reach the water ice ore deposit site, and what is the constituency of the water ice ore in the regolith? We are assuming that it has low-strength, fluffy physical properties[14] and is present in minimum concentration levels of ~10% by weight[3,9]. This is a critical factor in the mining and ore processing function of the outpost, and would impact the designs of those elements should the constituency and concentration change significantly from the assumptions. With those assumptions, our approach is to collect it with a bucket-loader, load it into a cargo bed that can be covered (to prevent sublimation), and haul it back to the Water Processor and Storage Package (WP&SP).

Depending upon the nature of the deposit, this concept can change from hauling out the feedstock to extracting the water *in situ* within the crater. However, at present, this is rather speculative and is why our first mission is a robotic prospector to determine the physical and chemical properties of the ice deposits. We have also adjusted the mass allocations for the WP&SP as well as the Water Electrolysis and Fuel Storage Package (WEFSP) to compensate for the increased propellant production needs of this architecture. We adjusted the power for the WEFSP as well to best quantify the power needed for electrolysis, using more up to date information on existing capabilities (C. Mittelsteadt, personal communication, 2016).

Because of the increased demand for propellant and the recognition that with so many arrays on the surface it may require periodic repositioning to prevent or minimize occultation, we have created new mobile power elements to augment the fixed Power Plant assets mounted to landers. The Mobile Power Package (MPP) is essentially vertical solar arrays on wheels, to move to the load location. The Lander Support Cart is created to solve a need to keep certain components on the reusable landers (RWTL and HL) warm and allow for cryogenic propellant storage, as well as to perform visual inspections on the landers or any other element on the surface. These are smaller than the MPPs and will need to move out of the takeoff/landing zones when the landers are used. These two mobile power units add more robust capability to the outpost and because of their mobility, can move to the load location to allow choreographing of the total power system at the outpost to better service the needs of the elements versus time.

We have also added increased functionality to the Water Tanker (WT) because of the realization that there will be significant need to move regolith and waste stream product. Therefore, since the duty cycle on the Water Tanker is low, we have added the capability for it to move regolith. Construction may include berms to essentially isolate the WEFSP (build a berm around the circumference) and the reusable landers to keep them cold-biased, to allow reduced cooling demand for cryogenic storage. Temperature extremes are a challenge to materials and thermal designs, and while the extremes are bad enough, the much more difficult challenge is repeated large temperature swings. Therefore, elements or subsystems that are designed to be cold should be cold-biased and not subject the direct solar radiation at any time. At the poles, this can be done by simply building a berm high enough so that the sun (which is never high above the horizon) is occulted from the element or part of the element in question. This will necessitate regolith movement to shape the terrain. As with all other roving elements, the WT operates telerobotically, with control initially from the Earth, transitioning to local control at the outpost when occupied by crew. We have also added Space Suits under the Human Power & Logistics Cluster, which was an omission in the previous architecture, although uncommitted mass allocations could have been used. The summary of surface elements, masses, and their functions is provided in Table 5.

Table 5. Surface Elements

Surface Elements: Rovers	
Characteristics	**Rationale**
	WATER ICE EXPLORER (WIE)
Purpose	To traverse into lunar polar craters and determine the properties and overall abundance and layout of water ice ore deposits at perspective outpost sites for the Lunar Outpost. One WIE will roam each pole at a preferred site. It is desired that the WIE is powered by an RTG, but if that is not available, the design will need to change to batteries, with the need to recharge. The WIE will have the necessary sensor suite, as well as a limited ability to "mine" water ice ore for future water processing. All locomotive movement is controlled telerobotically.
Mass	500 kg; based upon RLEP-2 studies
Operations Concept	Assuming an RTG power source, the WIE will land on a Robotic Medium Lander (RML), offload from the lander, and travel into the crater. Once there, it will investigate the water ore potential, gathering data on resource size, ease of mining, ease of access, traverse scheme, distance from a water processing element, and time to extract ore. Even with an RTG, batteries will be required for peak operations. This should allow for the WIE to remain in the crater most of the time, thereby minimizing temperature extreme cycles. Once the ore deposit has been investigated in a non-invasive manner, then the next phase will actually engage the deposit and attempt to capture a portion of it for transport back out of the crater.
Notes	If RTG's are not possible, then the WIE power and operational scheme will need to be redesigned around a vehicle that goes in and out of the crater (more challenging thermal environment), as well as having the need to plug into a power source at the lander (RML). The RML will need to increase in size. Note that Mission 1 does have TLI margin, so that if there is a need to grow mass to meet objectives, there is mass available with SLS Block 1.

Table 5. Surface Elements (continued)

Surface Elements: Rovers	
Characteristics	**Rationale**
	EXCAVATOR/HAULER (EH)
Purpose	To excavate and haul water ice ore feedstock from a cold crater to the Water Processor & Storage Package (WP&SP) in a protective environment to prevent sublimation, assuming a 10% wt ratio of water ice and a traverse distance of 10 km. Assume battery-powered, so that it would need to recharge after every excursion. All locomotive movement is controlled telerobotically. The EH Element may consist of distinct separate physical devices, if that is more efficient in design to meet the functional purpose. The EH will have at least 2 input ports to ingest power.
Mass	4 t; based upon LAT-1 studies, with factor 2 increase due to uncertainty
Operations Concept	EH is delivered to the Outpost via the RHL Lander, and drives off of the RHL under its own power. Normally parked at a charging station, the EH would unplug with a full charge, traverse the crater lip to the excavation site, and excavate enough to fill the Ore Carrier. Once full, the EH would return to the WP&SP, plug in at the WP&SP, and then dump ore/load the WP&SP. Once completed, the EH would return to the charging station for a full charge before starting again. The EH Element is commanded and controlled telerobotically. The charge process must be accomplished in less than 10 hrs, assuming that 14 hrs completes one ore cycle.
Notes	Many assumptions for the nature of the water ice ore deposits make this Element's concept speculative. The ore grade is assumed to be 10% wt water ice mixed with regolith and assumed to be light and fluffy in nature such that the EH can "scoop" up the mixture, load it, cover it, and drive back out of the crater. The EH will need to cold-bias the design, so that the EH does not see warm temps if at all possible. Seals must be able to work at very low temps, and regolith dust must be completely prevented from entering rotating joints. Need to lift the ore into the WP&SP either by lifting the bed (gravity assist) or conveyer or some other means (power for loading feedstock can be provided by an external Element. The EH element must also be designed to easily plug into a standard charging port anywhere on the Moon. All functions need not be contained on a single mobile Element; the ore hauling function can be a wagon. To produce 0.55 t water per day (average), and assuming a single trip into the crater per day, the hauling function must haul approx 6 t of ore out per day.
	WATER TANKER (WT)
Purpose	To load water from the WP&SP, transport to the WEFSP, and unload it to the WEFSP. In addition, it will also need to clear the waste stream from the WP&SP. It will also be responsible for building any berms and shaping terrain to isolate the landing area(s) from the rest of the Outpost. This element will be equipped with a robotic arm to help make electrical connections and to act as a portable camera when necessary. All locomotive movement is controlled telerobotically. It is powered by batteries; batteries are charged by an external power source. The WT will have at least 2 input ports to ingest power.
Mass:	
- Dry Mass	1 t
- Payload Mass	1 t (water)
Operations Concept	WT is delivered to the Outpost via the RHL Lander, and drives off of the RHL under its own power. Normally parked at a charging station, the WT will unplug with a full charge, travel to the WP&SP, and plug up a fluid line to flow water from the WP&SP. The water tank would be fully loaded, and then the fluid line would be disconnected and the WT would travel to the WEFSP. Once there, the fluid line would be connected to the WEFSP, and then water would flow into the WEFSP. Once complete, the WT would disconnect and return to its charging station. As a separate function, the WT would disconnect from the charging station at full charge and go to the WP&SP and would then push/remove the waste from the WP&SP. As a separate function, the WT would disconnect from a charging station and push/shape terrain as appropriate, telerobotically. When complete, it would return to the charging station.
Notes	The WT should be designed as a multi-function rover. The size of its water tank should be optimized between the WP&SP and the WEFSP water tank sizes and the operational concept and duty cycle of the other tasks especially WP&SP waste management. The WT must also be designed to easily plug into a standard charging port anywhere on the Moon for charging its batteries.
	ROVER FUELING TANKER (RFT)
Purpose	To load LOX and LH_2 from the WEFSP, transport to either the HL or the RWTL, and load propellants into those landers. All locomotive movement is controlled telerobotically. It is powered by batteries. The RFT will have at least 2 input ports to ingest power.

Table 5. Surface Elements (continued)

Surface Elements: Rovers	
Characteristics	**Rationale**
Mass:	
- Dry Mass	1 t
- Payload (Prop) Mass	1 t
Operations Concept	RFT is delivered to the Outpost via the RHL Lander, and drives off of the RHL under its own power. Normally parked at a charging station, the RFT would unplug with a full charge, travel to the WEFSP, and plug up both a LOX Line and a LH_2 line. The RFT would be fully loaded, then unplug and travel to either the HL or RWTL. Once there, it would connect up and load both LOX and LH2, one at a time. Once disconnected, the RFT would return to a charging station.
Notes	The WT should be designed as a multi-function rover as a backup for the WT. The size of its propellant tanks should be optimized between it and the WEFSP tank sizes and the operational concept and duty cycle. The RFT must also be designed to easily plug into a standard charging port anywhere on the Moon for charging its batteries.
	MOBILE POWER PACKAGE (MPP)
Purpose	To provide mobile power of 50 kW to the WEFSP or the WP&SP or any other user on a deployable/retractable vertical solar array mast. The MPP would move under its own power, discretely at different periods of time. The MPP would provide at least 4 separate input ports to supply power to loads. The MPP requires the array to have a single rotation axis normal to the ground to track the sun's progression around the horizon. Because of mobility, the MPP may have to retract arrays to a state that can be moved.
Mass:	1.9 t
Payload Power:	50 kW
Operations Concept	MPP is delivered to the Outpost via the RHL Lander, and drives off of the RHL under its own power. The MPP would traverse to the load, move to a favorable solar viewing position, and connect electrically to the appropriate Element (i.e., WEFSP). Once connected, the MPP would stabilize if necessary, and deploy a vertical mast and unfurl the solar array. When it is time to move again, the process would be reversed; furl the array, stow the mast, collect any cabling, and unplug and either move to a different viewing spot with the same load or traverse to supply a new load.
Notes	Even though the concept of providing mobile power with retracting arrays is more complicated than fixed packages, there will be a need to manage and orchestrate power around the Outpost due to Outpost and terrain occultations that will occur from time to time. To allow for this, the MPP was conceived. To prevent tip-over, it might be required to deploy horizontal stabilization legs when arrays are deployed.
	LANDER SUPPORT CART (LSC)
Purpose	To provide mobile power to the HL and the RWTL Elements for all lander ground functions, including cryo cooling to prevent boiloff after cryo loading. LSC will need to disconnect and move away from the launch site when the landers are taking off and landing. The LSC may also need to move while connected to optimize solar viewing. As a secondary function, the LSC can power and recharge any element on the surface. All locomotive movement is controlled telerobotically. The LSC would provide at least 3 separate input ports to supply power to loads. The LSC requires the array to have a single rotation axis normal to the ground to track the sun's progression around the horizon. Because of mobility, the LSC may have to retract arrays to a state that can be moved.
Mass:	0.5 t
Payload Power:	10 kW

Table 5. Surface Elements (continued)

Surface Elements: Rovers	
Characteristics	**Rationale**
Operations Concept	LSC is delivered to the Outpost via the RHL Lander, and drives off of the RHL under its own power. Once at the Moon, the LSC will travel to either an HL or a RWTL and plug into the lander. Once plugged in, it will deploy solar arrays and provide power to the lander. When the lander is ready to take off, the LSC will retract the arrays, unplug from the lander, and move a safe distance away. Once the lander is off the surface, the LSC can move to another element and plug in and unfurl solar arrays while it is waiting for the lander to return. However, once a lander has landed, the LSC must connect up quickly to prevent loss of propellant through boiloff. Since the LSC may be connected to a lander for a significant period of time, the LSC may need to physically move to a more advantageous solar orientation from time to time to minimize eclipses.
Notes	It might be worth investigating adding capability to the basic LSC platform for robotic arms or enhanced communication or other need not fully realized at this point. To prevent tip-over, it might be required to deploy horizontal stabilization legs when arrays are deployed.
	Surface Utility Vehicle (SUV)
Purpose	To provide mobile crew transport in a shirt-sleeved environment including shirt-sleeved ingress/egress for the HL and Habitats. It will also provide airlock ingress/egress.
Mass:	4.5 t
Crew:	4
Operations Concept	SUV is delivered to the Outpost via the CL Lander, and drives off under its own power. It would plug into a spare port in either a fixed PP or MPP as appropriate. When needed, it would unplug, move to the HL that has just landed, and dock to the HL to receive crew. The crew would transfer and then undock, maneuver to the Habitat, dock, and then transfer crew.
Notes	Need for this to be as small and as utilitarian as possible; duty cycle is such that it is only used for short periods of time. Trade need for robotic arms to help outside tasks in addition to crew mobility.
Surface Elements: Permanently Mounted Payloads	
	Habitat
Purpose	To provide living and inside working quarters for a crew of 4. The Habitat would have an Airlock and a port for shirt-sleeve ingress/egress with the SUV. Current design is that the Habitat is permanently attached to the CL.
Mass:	10 t
Crew:	4
Operations Concept	The Habitat essentially lands with the CL and is fixed to the lander. The SUV comes to the Habitat for Crew exchange.
Notes	Future trades would be to figure out how to expand the habitable volume in a contiguous manner to allow growth, yet this would require transport of additional habitable volumes to move to the Habitat or vice versa. In addition, study should be made of inflatables and using local processed regolith to build habitat structures. Examine leverage potential of Constellation work with Commercial suppliers for Inflatable/Expandable Habitat Structures.

Table 5. Surface Elements (continued)

Surface Elements: Rovers	
Characteristics	**Rationale**
	POWER PLANT (PP)
Purpose	To provide 75 kW of power to various loads in a fixed configuration (assuming initial deployment). It would require a single rotation axis normal to the ground to track the sun's progression around the horizon. As well, it should have at least 3 generic power charging ports to support load inputs.
Mass	1.1 t; based upon LAT-1 studies
Power	75 kW (24 kW-hrs keep-alive power in eclipse via 200 kg of batteries)
	POWER PLANT (PP) - Continued
Operations Concept	The PP will be hard-mounted to the RHL and would be unfurled once the RHL lands. It could be first a mast extension vertically, followed by an unfurling of the array itself. Because of the shortage of battery life, this will be one of the first things done once landed.
Notes	Need to trade the shape of the array for best fitting optimum need. It seems like vertical rectangular arrays would give most flexibility, but circular versions might be most effectively packaged.
	WATER PROCESSING & STORAGE PACKAGE (WP&SP)
Purpose	To process water ice ore provided by the EH element by stripping out the water and discarding the waste. The WP&SP will also store water after processing in liquid form and interface with the WT for water transfer.
Mass	3 t; based upon LAT-1 studies
Output Production	170 t water per year
Operations Concept	Once at the Moon, the WP&SP would be fixed to the RHL along with the 75 kW PP. When the EH unit is loaded, it will bring the ore load to the WP&SP and transfer it (lifting bed on EH to trade against conveyer on WP&SP). From there, the WP&SP would heat (electric or collector mirror) the ore above the water melting point. Assume a batch of 6.5 t ore per day, producing 650 kg water/day. When ready, the WT rover would connect up to the WP&SP and the WP&SP would transfer the water to the WT for transport.
Notes	Design to minimize threat of sublimation of water; may need to contain all vapors. Also, if there is any additional processing for other species, this is future work. Note that a different element (WT) will remove the waste pileup that will build up over time due to the size of the waste stream.
	WATER ELECTROLYSIS & FUEL STORAGE PACKAGE (WEFSP)
Purpose	To electrolyze lunar-produced water provided by the WT element and produce both GOX and GH2, and then liquefy both gases into LOX and LH_2 storage tanks. The WEFSP will receive its power from both a fixed PP as well as an MPP. The WEFSP will interface with the WT to transfer water into the WEFSP, and will also interface with an RFT to transfer both LH_2 and LOX to the RFT for use or storage elsewhere.
Mass	2 t; based upon LAT-1 studies extrapolated to reach output rates
Output Production	140 t/yr of processed water (15.5 t H_2, 124.5 t O_2)

Table 5. Surface Elements (continued)

Surface Elements: Rovers	
Characteristics	**Rationale**
Operations Concept	Once at the Moon, the WEFSP would be fixed to the RHL along with a fixed power source, a 75 kW PP. An MPP would maneuver over and connect to the WEFSP and deploy arrays with an additional 50 kW power for electrolysis and liquefaction and system loads. When the WT arrives, it transfers 1t of water to the unit, and the unit electrolyzes the water into GH_2 and GO_2; both gases are then cooled and liquefied, and kept stored in the liquid state. When appropriate, the RFT will transfer the liquid propellants to the landers for use or storage, allowing the WEFSP to keep the storage size small.
Notes	The design will be a challenge to keep the water electrolysis region hot, while keeping the liquefaction region cold. To do this, it might be necessary to have those functions physically separated; maybe have the liquefaction function on a mobile system that moves into a sheltered (berm) area to keep very cold. Assumes 80% sunlight per year, 100 kW applied to electrolysis

XI. Mission Sequence

As with the earlier architecture[2,3], the first mission is a robotic exploration mission to measure the nature of the lunar ice deposits and map their extent, as this is critical information to design hardware elements to extract the ore. Thus, the first part of the architecture has not changed. From that point forward, we build the outpost using the SLS and the RHL as the main transportation element with a single launch, direct to the lunar surface. These launches will not be addressed in detail, as they are straightforward, using the Block 1 SLS. This section will go into further detail regarding crew rotations. The first element in the crew rotation chain is the lunar HL. The SLS Block 1B with an EUS launches a fully fueled (but uncrewed) HL. The EUS performs the TLI burn and is then discarded. The HL performs the LOI, then the LLO circularization, and then rendezvous with the LLO Depot, to stage for the future crew mission. Once docked to the LLO Depot, the HL will stay there until the CCS arrives in LLO with crew. (A future trade can examine the benefit of landing an uncrewed HL to the surface and then automatically refueling and launching from the outpost when ready to receive the crew in LLO.)

The second element is the Cislunar Crew Stage (CCS). The SLS Block 1 with no upper stage launches an uncrewed CCS to LEO orbit with enough propellant to rendezvous and dock to the LEO Depot. The CCS will rendezvous and dock to the LEO Depot and wait for the first crew rotation while filling its propellant tanks. Previously, the depot will have been filled with water from commercial launches. The first crew rotation starts with a launch by the CCCrew program with 4 crew to rendezvous and dock with the LEO Depot while the CCS is docked. At the right time (typically just a day or two), the crew will transfer from the depot to the CCS, undock, and then perform TLI to the Moon and then perform LOI burns to rendezvous with the LLO Depot. To accommodate LLO depot orbit phasing, the CCS will perform LOI in three phases, first into an intermediate elliptical orbit around the Moon with a high apoapse, later a plane change at apoapse to line up with the LLO Depot orbit, then followed by the final burn to enter the LLO Orbit. The HL will be waiting for the CCS at the LLO Depot. The CCS will perform the rendezvous and docking with the LLO Depot. The crew will transfer into the LLO Depot and dwell for a few days if there are any outpost orientation constraints for the first few landings. When the outpost landing strip orientation is lined up with the LLO Depot (or other landing constraints satisfied), the crew will transfer from the LLO Depot to the HL, and then the HL will undock and deorbit and descend to the outpost. Once landed, the Surface Utility Vehicle (SUV) will drive to the HL and dock to it to allow shirt-sleeve egress from the HL into the SUV. Once complete, the SUV will undock from the HL drive to the Habitat, dock, and transfer the crew. The SUV will stay connected to the Habitat while the crew is in the Habitat.

Meanwhile, the CCS will refuel for the return journey to LEO. The uncrewed CCS stays at the depot for a day or two for fueling as well as orbit phasing, and then leaves the LLO Depot and performs the first TEI burn into another interim elliptical orbit to allow a more efficient plane change burn at apoapse, followed by the final TEI burn to travel to Earth. Once at Earth, the CCS will use aerobraking to demonstrate the technique without crew and to ensure that it can be done safely. Once the aerobraking has successfully shed the appropriate DV using the two-step orbital pass maneuver, the CCS will rendezvous and dock to the LEO Depot and refuel while waiting for the next crew. The first CCCrew capsule will already be docked at the LEO Depot in a dwell state.

For every crew rotation from that point forward, the crew cycle will start with a CCCrew launch of 4 incoming crew to the LEO Depot while the returning crew is still at the outpost. The incoming crew will transfer into the depot, and then when ready (a day or less dwell) they will transfer into the CCS for the forward journey to the Moon, using the same sequence as defined above. The returning crew meanwhile will phase their HL trip to the

LLO Depot in time so that they are at the depot when the new crew arrives. The CCS with the new crew will dock to the LLO Depot and the new crew will transfer into the LLO Depot while the returning crew is also in the depot. Assuming no landing constraints, the new crew can transfer into the HL, undock, and land at the outpost at any time. The returning crew will transfer into the CCS after the CCS refuels, and then the CCS will leave the depot in favor of the TEI phasing orbit, followed by the TEI burn to bring the returning crew to Earth. Once in the Earth vicinity, the CCS performs the two-pass aerobraking and then maneuvers to the LEO Depot orbit to link up with the LEO Depot. The returning crew can then transfer into the LEO Depot and into the previous CCCrew capsule, docked at the LEO Depot for 6 months, at any time for return to Earth.

The HL, meanwhile, will land the new crew at the outpost. As before, the SUV will drive up to the HL and the crew will transfer into the SUV and drive to the Habitat to start a new rotation. After landing, the Lander Support Cart (LSC) will drive over to the HL, plug in power, and unfurl arrays to keep the HL healthy. At various points prior to the next crew rotation, the Rover Fueling Tanker will drive to the HL and refuel. The LSC will provide sufficient power to minimize LH_2 boil-off while the HL dwells on the surface. To minimize boil-off, the tank portion of the HL may need to be shielded from the sun by a circumferential berm so that the tanks stay in shadow to ease the cooling burden on the tanks for the long dwell times. Note that our new architecture features near-continuous human presence on the Moon whereas previously, we were only occupying the lunar surface in sequential three-month segments (alternating 3 months inhabited, 3 months uninhabited).

This sequence allows a fully independent and redundant crew launch capability, with reusable elements. For emergency return, the Orion will be docked at the LLO Depot as a lifeboat, capable of returning one or more crew directly to the Earth. Our timeline has a second Orion lifeboat every 5 years, as well as a second RWTL for redundancy and lifetime change out due to the stricter design parameters and higher usage on the RWTL. We did not include budget or manifest schedule for a second HL or CCS. To maximize operational reliability, both human reusable elements (HL and CCS) will have single engine out capability as well as engine change out capability. Future work on the concept can trade the cost versus benefit of launching a second HL and CCS during the phase 1 architecture just in case there is a problem that results in downtime or to add additional capability.

Using this new scheme with SLS and with CCCrew, we allow more crew flights with more crew rotations (36 crew vs. 20 crew) in 16 years. This alone offers a significant benefit over the previous architecture, since lunar surface crew time can be used in exchange for complementary elements in the architecture. Since this entire architecture hinges upon access to lunar water, our goal is to over-produce in the lunar water supply chain. In so doing, we enable an increase in crew rotation rates, with the real limiting factor being lunar water. While it is true that part of the fueling is done from the LEO depot with Earth-supplied water, we feel that this is not the dominant key variable setting the crew rotation rate. Any future consideration for this architecture should focus on producing more lunar water for Earth return propellant.

One final note on mission sequencing: because we want the CCS, HL, and RWTL to be designed to be most efficient structures with the lowest possible DMFs, it may be worth trading mission concept to fuel at the LEO Depot (instead of the present concept of launching fully fueled from the SLS) for each of the three elements if that element's launch load with propellant tanks mostly empty can appreciably lower the DMF. On the Moon and in-space for the CCS, we suspect that loads are considerably less, and so if the structural design case (most stressing condition) consists of fully loaded fuel tanks during liftoff and ascent, it might be necessary to change the concept to initially fuel at the LEO Depot after launch for each element so that the reusable elements can be as mass-efficient as possible.

XII. Cost and Schedule

Costs are summarized in Table 6. As in the previous version of the architecture[2,3], we estimate that a fully functioning lunar outpost – capable of producing ~ 500 tonnes of water per year and roughly 500 tonnes of propellant (this is capacity, not actually produced) – can be established with 4 crew stationed there continuously (rotating every 6 months) for an aggregate cost of roughly $87.7 billion (Real Year dollars), with a total of 216 crew-months lunar dwell time. This compares to only 120 crew-months dwell time for the previous architecture, at essentially the same cost ($550 million more (<1%) for this new architecture is within cost estimation noise). Similar to the previous version of the architecture, the outpost is deployed robotically and operations are fully implemented within 10-15 years of program start, but as the use of robotic assets early in the program makes the schedule flexible, we can either accelerate or slow the progress of the program, as fiscal circumstances require. Human arrival comes relatively late in the process (although 1 year earlier than the previous version), after we have established a productive resource processing facility but within a few years of the arrival of robotic surface assets.

As part of this iteration of our architecture, we maintained the same overall cost-capped profile that we had proposed[2] in 2010, with about $7 billion per year allocated to the lunar outpost program. Due to the nature of a program's development curve, there are 3 years that come in a little above \$7 billion/year. While a few of the year-to-year costs compared to the 2010 architecture[2] have increased, the total costs for this architecture have been preserved to be under the total cost of the previous one. One important note is that for the SLS and launch complex, CCCrew, and Orion, we do not include the development costs but have accounted for recurring fixed and variable costs for these items.

To take better advantage of the freedom to remove SLS and Commercial Crew development cost, we applied a closer look at the cost allocations of the remaining elements, as well as embellishing the smaller logistical concerns that were not addressed previously. We have now fully estimated those costs, including the Mobile Power Platforms (MPP) and the Lander Support Carts (LSC) as examples of more completely canvassing the architecture logistical needs and improving operational efficiency. The landers were increased in size and cost as well as the lunar propellant processing chain elements to accommodate the increased need for lunar propellants. There were additional savings in this architecture mentioned previously regarding the crew rotation cycle cost reductions, and these were fully incorporated.

In this version of the architecture, to enable the overall discussion with potential international partners, we suggest a number of items on the overall hardware element list that could be contributed elements. Some would be

Table 6. Architecture Costs

Mission	Description	Launch Vehicle	Lander #	Yr 1	Yr 2	Yr 3	Yr 4	Yr 5	Yr 6	Yr 7	Yr 8	Yr 9	Yr 10	Yr 11	Yr 12	Yr 13	Yr 14	Yr 15	Yr 16	Total
1	Lunar Comm & Bi-Polar WIE rovers	SLS Block 1	RML 01, 02	150	350	600	500	300												1900
2	Water Processor #1	SLS Block 1	RHL 001	200	450	800	800	650	420											3320
3	Water Tanker & MPP #1	SLS Block 1	RHL 002		50	150	200	250	135											785
*	DONATED COST: Water Tanker				50	150	200	200	100											700
4	Ore Excavator/Hauler #1	SLS Block 1	RHL 003			50	75	150	75											350
*	DONATED COST: EH #1			50	100	200	400	350	200											1300
5	Water Electrolysis #1	SLS Block 1	RHL 004			100	250	400	350	150										1250
6	Rover Fueling Tanker & Lander Carts	SLS Block 1	RHL 005				50	75	100	50										275
*	DONATED COST: RFT						25	75	100	50										250
*	DONATED COST: Lander Supt Carts						50	100	100	50										300
7	Reusable Water Tank Lander	SLS Block 1B	RWTL #1		50	250	400	450	500	350	200									2200
8	LLO Depot	SLS Block 1B		50	200	400	550	650	750	550	250									3400
9	Uncrewed Orion to LLO	SLS Block 1	Orion 1									0								0
10	Human Power and Logistics Cluster	SLS Block 1B	CL 01					50	350	550	500	250								1700
*	DONATED COST: Personnel Trans Vehicle						100	200	400	600	500	200								2000
*	DONATED COST: Comm Terminals								10	30	40	20								100
*	DONATED COST: LHRS Heavy (Cranes)								10	30	40	20								100
11	Water Electrolysis #2	SLS Block 1	RHL 006					50	100	150	180	100								580
12	Human Habitat #1	SLS Block 1B	CL 02							15	50	95	40							200
*	DONATED COST: Habitat #1							150	450	550	700	750	400							3000
13	LEO Depot	SLS Block 1							200	300	400	350	250							1500
14	Human Lander (reusable)	SLS Block 1B	HL 01					100	400	500	700	750	800	850						4100
15	Cis-Lunar Crew Stage	SLS Block 1B	CCS					50	200	600	800	700	500	250						3100
16	**First Human Mission**	**CCCrew**													500					500
17	Ore Excavator/Hauler #2	SLS Block 1	RHL 007									50	75	75	50					250
*	DONATED COST: EH #2											50	100	100	50					300
18	**Human Mission 2**	**CCCrew**													500					500
19	Water Electrolysis #3	SLS Block 1	RHL 008								50	100	150	180	100					580
20	Water Processor #2	SLS Block 1	RHL 009									100	150	130	100					480
21	**Human Mission 3**	**CCCrew**														500				500
22	Human Habitat #2	SLS Block 1	CL 03										5	15	30	20				70
*	DONATED COST: Habitat #2												100	300	350	250				1000
23	Water Electrolysis #4	SLS Block 1	RHL 010									50	100	150	180	100				580
24	**Human Mission 4**	**CCCrew**															500			500
25	Uncrewed Orion to LLO	SLS Block 1	Orion 2														0			0
26	Water Upmass Wedge	SLS Block 1	RHL 011											15	25	35	25			100
*	DONATED COST: Water Upmass Wedge												100	200	650	600	350			1900
27	**Human Mission 5**	**CCCrew**															500			500
28	Water Electrolysis #5	SLS Block 1	RHL 012										50	100	150	180	100			580
29	**Human Mission 6**	**CCCrew**																500		500
30	Water Processor #3	SLS Block 1	RHL 013												100	150	130	100		480
31	Reusable Water Tank Lander #2	SLS Block 1B	RWTL #2												100	200	200	150		650
32	**Human Mission 7**	**CCCrew**																500		500
33	Ore Excavator/Hauler #3	SLS Block 1	RHL 014													50	75	75	50	250
*	DONATED COST: EH #3															50	100	100	50	300
34	**Human Mission 8**	**CCCrew**																	500	500
35	ISRU Wedge	SLS Block 1	RHL 015													15	25	35	25	100
*	DONATED COST: ISRU Wedge														100	200	650	600	350	1900
36	**Human Mission 9**	**CCCrew**																	500	500
																				0
	Block 1B																			
	SLS			1000	1200	1400	1500	1640	1750	1870	1995	1970	1850	1875	1800	1820	1775	1675	1600	26720
	Orion (includes non-lunar ops)			200	200	200	200	200	250	300	350	300	200	250	300	350	300	200	200	4000
	Ground Systems at KSC			200	200	200	200	200	300	300	300	300	300	300	300	300	300	300	300	4300
*	DONATED: Cargo Lander DDTE & Production						150	300	400	500	500	400	250	150	200	150				3000
	Engine Technology Wedge			300	600	600	300													1800
	Architecture Integration (lunar ops)			50	60	70	80	90	100	100	100	100	100	100	100	100	100	100	100	1450
	Totals per year			2200	3510	5170	6030	6680	7750	7595	7655	6655	5520	5040	5685	5070	5130	4335	3675	87700
*	NOTE: Asterisk (Blue lines) are suggested International or Commercial Partners																			
	Assuming contributions shown in blue, year to year cost to NASA:			2150	3260	4670	4855	5015	5530	5515	5530	4895	4320	4215	4335	3750	3955	3560	3275	68830
																			Total Donated:	16150

in the critical path toward human outpost initial operational capability. The net result is that if there are contributed elements (which could be bartered for additional lunar surface crew time in this architecture), the total cost would be about $18 billion less for this architecture as estimated for this study if our suggestions for contributed elements were adopted. *Without any contributed elements*, the cost for the 2016 version would be roughly the same as the 2010 version at the granularity of this study. The cost for the SLS launch elements, launch infrastructure, CCCrew seat cost, Orion cost, and operations costs were determined with published budget information and cost extrapolations based upon experienced knowledge of NASA systems in general.

XIII. Architecture Validation

We have purposely developed a logical, cogent explanation for the architecture choices that we have made, first anchored in an overarching goal to "Expand human reach beyond Earth, while opening the human economic sphere using off-planet resources." We have determined the best destination to execute the goal and defined a set of objectives and scoped and matured an architecture that meets those objectives. While we admit that there are many ways to implement our objectives, this architecture seems to be closely aligned with existing or soon to be existing capabilities of both the SLS and CCCrew development programs, and therefore to us represents an effective and simple architectural concept that maximizes and leverages previous agency investments and already developed assets while meeting all of our goals and objectives.

While our architecture addresses Phase 1 of the objectives, it enables the pursuit of the Phase 2 objectives, which examine the more long-term viability of human presence off-planet. These are the objectives that will determine if human expansion is truly viable or simply an expensive distraction if it is shown that we cannot effectively use planetary resources to support human endeavors in space. Our opinion is that there is great potential for exponential success, but we should pay the significant startup costs to provide definitive answers.

XIV. Conclusions

We believe that the inevitable next step in human exploration is to extend human reach in space beyond its current limit of low Earth orbit through the use of local *in situ* resources. The Moon has the material and energy resources needed to examine the viability of resource utilization to first and foremost establish a human transportation route to and from the Moon and consequently, throughout cislunar space. Data show that the lunar surface is rich in resource potential; both abundant water and near-permanent sunlight are available at selected areas near the poles. We go to the Moon to learn how to extract and use those resources to create a space transportation system that can routinely access all of cislunar space with both machines and people. We also go to the Moon to learn how to live off-planet by exploiting its material and energy resources in an ever expanding sphere of capability, first to determine if it can be done, and then to examine the practicality and viability of such a unique yet untried approach. Such an audacious goal makes our national civil space program relevant to economic and national security interests as well as to scientific ones, since there will be significant science to be performed wherever we go.

This return to the Moon is affordable based upon past costs of long-lived space programs. We believe that the US must once again exert leadership in space, which has come into question in the last several years as we appear to be adrift from lack of a human spaceflight purpose. We provide such a bold plan, with a logical set of objectives, and an implementation to meet them.

Our architecture concept can be adjusted in schedule to fit any monetary or programmatic shortfall. But regardless of program pace, our goals and tactics remain the same: to open the space frontier for a wide variety of purposes by harvesting the material and energy resources of the Moon and then examining the true potential of use of *in situ* resources to thrive off-planet. Now is the time for boldness and determination to reach forward and outward. Let us not waste this opportunity any longer.

"If God wanted man to become a space-faring species, He would have given man a Moon." – Krafft Ehricke, 1985[14]

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