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Impact of the Lunar Gateway Location on the Human Landing System in case of Permanent Base at the Lunar South Pole

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Abstract

NASA's Gateway platform will be a critical element in enabling long duration crewed missions on the Moon. The deep space station location will define the trajectories for transfers between the Earth and the Moon and thus affect the lunar transportation architecture. One of the most affected systems will be the human landing system (HLS), whose main function is to deliver the crew from the Gateway to the lunar surface and back.

This paper discusses the impact of the Gateway location on the HLS elements sizing. In total, 12 Gateway orbits are considered, including polar circular and elliptic lunar orbits of different sizes, near rectilinear halo orbits (NRHOs) of L1 and L2 type, and an L1 conventional halo orbit. As the issue at hand involves crewed missions, the effect of potential abort operations (which differ for different Gateway orbits) on the resultant element sizing is also included in the analysis. NASA's 3-stage architecture, which includes descent, ascent, and transfer vehicle elements, is adopted for the HLS. Rocket equation and mass estimating relationships are used to size the HLS elements for different Gateway orbits.

Our analysis has revealed that all three elements are affected by the Gateway location. NRHOs, which are NASA's current baseline option for the Gateway location, are among the orbits with highest wet mass of all HLS elements. The lightest HLS corresponds to the Gateway in a 100-km polar circular orbit. In this case, the transfer vehicle is not needed at all while the descent and ascent elements are 23% and 26% lighter respectively than their NRHO counterparts. This increase in the HLS mass along with more restrictive abort operations is a drawback to the benefits that a Gateway in NRHO can potentially provide.

Keywords: human landing system, Lunar Gateway

Acronyms/Abbreviations

HLS – human landing system
NRHO – near rectilinear halo orbit
GW – Gateway
LLO – low lunar orbit
LLAO – low lunar arrival orbit
LLDO – low lunar departure orbit
LLAbO – low lunar abort orbit

1. Introduction

The space community is actively preparing for the next step in human space exploration which involves establishing a sustainable human presence on the Moon, with the first lunar outpost being presumably located at the lunar South Pole [1].

The current NASA lunar exploration architecture includes a habitable deep space station in cis-lunar space – the Gateway. The Gateway will be a critical element in enabling long-duration crewed missions to the Moon, serving as a central location for aggregation of resources and supplies for human lunar missions and a staging point for transportation systems transferring people and cargo between the Earth and the Moon.

The Gateway station location will define the trajectories for transfers between the Earth and the Moon and thus affect the lunar transportation architecture. One of the most affected systems will be the human landing system (HLS) whose main function is to deliver the crew from the Gateway to the lunar surface and back. This paper explores the impact of the Gateway location on the HLS elements sizing.

There are a number of potential locations for the Gateway station including low lunar orbits, halo orbits near Earth-Moon libration points L1 and L2, near rectilinear halo orbits, and some others. Those orbits have been analyzed and compared to each other using such criteria as delta-V costs of accessing those orbits from the Earth and the Moon, station-keeping costs, shadow avoidance, communication with Earth and lunar landing sites, and extensibility to future Mars missions [2-4]. However, specific impact of the Gateway location on the performance of individual transportation systems has not been explored. This paper is an attempt to address this gap for human landing systems. In total, 12 different Gateway orbits are considered including:

- 3 circular polar orbits of altitudes 100 km, 500 km, and 5000 km;
- 6 elliptic polar orbits (100x5000 km, 100x10000 km, 100x20000 km, 1000x5000 km, 1000x10000 km, 1000x20000 km);
- 2 near rectilinear halo orbits (L1 11:3 Southern NRHO and L2 4:1 Southern NRHO);
- 1 L1 halo orbit (z-amplitude = 12000 km, Southern type).

A single landing site (a permanent lunar base) at the lunar South Pole is assumed.

To date, three HLS architectures have been proposed for the Gateway-supported lunar infrastructure: 1-stage, 2-stage, and 3-stage. A 1-stage HLS performs all transportation operations without leaving any of its elements on the lunar surface; it is potentially fully reusable. The 2-stage architecture uses the expendable descent stage to deliver the ascent stage to the lunar surface and the ascent stage to return the crew back to the station; only the ascent stage can be reused in the subsequent missions in this case. The 3-stage architecture uses the transfer vehicle to deliver the descent/ascent stack to a descent low lunar orbit (LLO), the expendable descent stage to deliver the ascent stage from the descent LLO to the surface and the ascent stage to return the crew back to the station; in this case, two HLS elements – the transfer vehicle (which returns to the Gateway from the descent LLO on its own) and the ascent stage – are potentially reusable. Recently, NASA selected three companies to develop human landing systems for its Artemis program [5]. Two of them, Dynetics and SpaceX, are developing 1-stage systems; the third one, Blue Origin, is working on a 3-stage system following the NASA's design reference used by the agency for soliciting commercial proposals [6, 7]. For our study, we assumed the 3-stage HLS architecture with all three HLS vehicles using LOX/LH2 propulsion.

Section 2 describes the HLS concept of operations and parametric model used for sizing the system's elements. Section 3 analyzes impact of the Gateway location on the HLS elements mass and operations. Limitations of the analysis and potential further work in this area are discussed in Section 4. Section 5 provides a brief summary of the study findings.

2. Methodology

A combination of the rocket equation expressions yielding from the HLS concept of operations and the respective delta-V requirements and parametric mass estimating relationships for individual HLS elements was used to derive the HLS sizing equations. Subsection 2.1 describes the concept of operations and the delta-V requirements adopted for sizing. Subsection 2.2 discusses the 3-stage HLS parametric model and the resulting sizing equations.

2.1 HLS Concept of Operations and Delta-V Requirements

The following nominal concept of operations was assumed for the 3-stage HLS:

1. The lunar mission starts at the Gateway, where the assembled and fully fuelled HLS, loaded with the payload to be delivered to the lunar surface, awaits for the crew arrival.
2. When the mission starts, the crew residing at the Gateway transfers to the HLS ascent stage. The HLS undocks from the station, and the transfer vehicle propels it towards the Moon. Upon the arrival, it injects the HLS into a 100-km polar low lunar orbit (the low lunar arrival orbit (LLAO)) which contains the designated landing site in its plane. The transfer vehicle then undocks from the descent/ascent stack and returns to the Gateway on its own.
3. The descent vehicle delivers the stack from the LLAO to the designated landing site at the lunar South Pole, performing an in-plane descent. The descent element is now fulfilled its primary function and serves as a launch pad for the ascent vehicle for the rest of the mission. While on the surface, the crew lives in the ascent vehicle. During the surface operations, the payload delivered from the Gateway is unloaded to the surface or left in the descent stage; the payload to be returned to the Gateway is loaded to the ascent vehicle.
4. After the surface operations are over, the ascent vehicle with the crew onboard performs an in-plane ascent from the landing site to another 100-km polar low lunar orbit (the low lunar departure orbit (LLDO)) leaving the descent stage behind. The vehicle then departs the LLDO for the Gateway.
5. At the Gateway, the crew and the payload returned from the lunar surface are transferred to the Gateway.

Since HLS is used for transporting humans, apart from nominal transfers, it should also be able to perform abort transportation operations which often use trajectories different from the ones for nominal transfers. When defining the delta-V requirements to be used for sizing the HLS elements, we additionally considered two abort scenarios:

- Abort from the LLAO is initiated if an emergency occurs that prevents the subsequent descent to the surface after the HLS has arrived at the LLAO. In this case, the transfer vehicle performs the nominal sequence undocking from the HLS and returning back to the Gateway on its own. Depending on the character of emergency, either the descent or the ascent vehicle performs the propulsive burns needed to return the crew to the station on the respective abort trajectory. If the ascent vehicle is used for that purpose, it needs to undock from the descent stage before it departs from the LLAO.

- Abort from the lunar surface is initiated if an emergency occurs at any time on the surface that requires the crew to urgently leave the Moon. In this case, the ascent vehicle is used for the abort return which includes two operations: an in-plane ascent into a 100-km polar low lunar abort orbit (LLAbO) and the subsequent transfer from this orbit to the Gateway on the respective abort trajectory. Depending on the character of the emergency and the time of abort, the ascent vehicle might have to loiter in the LLAbO prior to its departure.

Table 1 contains the delta-V requirements allocated to the individual HLS vehicles based on the concept of operations outlined above, for the 12 Gateway orbits considered. Table 2 contains data on the respective transfer times as well as surface access/departure frequencies and optimal surface stay times for the same orbits (for nominal operations, it was assumed that the surface access and departure windows as well as the surface stay time are selected to minimize the mission's total nominal delta-V cost). The details on how these values were obtained can be found in [8].

2.2 HLS Parametric Model

Ref. [9, 10] provide simple mass estimating relationships for human descent and ascent vehicles derived from the Apollo data for conceptual studies, and a similar relationship for orbital transfer vehicle. We used those relationships to model the individual HLS elements of the 3-stage HLS adapting them to the case at hand, where necessary.

Apollo-derived ascent vehicle model. The ascent vehicle is assumed to include a crew cabin and a propulsion module and to also carry consumables (such as food and water for the crew and ECLSS consumables to compensate for leakages/crew metabolic needs). The dry mass of the vehicle is then estimated as follows:

$$m_{dry}^{asc} = m_{cabin} + m_{consumables} + m_{dry\,PM}^{asc} \tag{1}$$

Ref. [10] provides the following estimate for the mass of the crew cabin which is assumed to include structure, ECLSS hardware (including the airlock), crew and crew provisions (such as seats and suits):

$$m_{cabin} = 1250 + 525 n_{crew} \tag{2}$$

Here, n_{crew} is the number of crew supported by the ascent vehicle. The data provided in [10] yields the following equation for the mass of consumables needed per mission:

$$m_{consumables} = 9.4 n_{crew} t_{support} + 2.3 t_{support} + 4.5 n_{EVA} \tag{3}$$

Here, $t_{support}$ is the total number of days the crew cabin provides life support to the crew (including in-space transportation and surface stay); n_{EVA} is the number of extravehicular activity cycles performed by the crew on the surface. The dry mass of the LOX/LH2 ascent vehicle propulsion module (in kg) is estimated as follows [9, 10]:

$$m_{dry\,PM}^{asc} = A_a m_{launch} + B_a m_p^{asc} + C_a \tag{4}$$

Here, $A_a = 0.064$, $B_a = 0.1653$, $C_a = 390$ kg are the model coefficients; m_p^{asc} is the ascent vehicle propellant mass; m_{launch} is the total wet mass being launched from the lunar surface. For the adopted concept of operations, the latter is calculated as follows:

$$m_{launch} = m_{dry}^{asc} + m_p^{asc} + m_{PLD}^{up} \tag{5}$$

Here, m_{PLD}^{up} is the payload mass to be returned from the surface. The wet mass of the ascent vehicle is defined as follows:

$$m_{wet}^{asc} = m_{dry}^{asc} + m_p^{asc}$$

Apollo-derived descent vehicle model. The descent vehicle consists of the propulsion module only. Its dry mass (in kg) is estimated similarly to the dry mass of the ascent vehicle propulsion module [9, 10]:

$$m_{dry\,PM}^{dsc} = A_d m_{deorbit} + B_d m_p^{dsc} + C_d \tag{6}$$

Here, m_p^{dsc} is the descent vehicle propellant mass. The model coefficients for the LOX/LH2 descent stage are the same as for the ascent stage propulsion module: $A_d = 0.064$, $B_d = 0.1653$, $C_d = 390$ kg. For the adopted concept of operations, the total wet mass being de-orbited prior to descent to the lunar surface $m_{deorbit}$ is defined as follows:

$$m_{deorbit} = m_{dry}^{dsc} + m_p^{dsc} + m_{dry}^{asc} + m_p^{asc} + m_{PLD}^{down} \tag{7}$$

Here, m_{PLD}^{down} is the payload mass to be delivered to the surface. The wet mass of the descent vehicle is defined as follows:

$$m_{wet}^{dsc} = m_{dry}^{dsc} + m_{p}^{dsc}$$

Transfer vehicle model. The following mass estimating relationship for orbital transfer vehicles [9] was used to estimate the dry mass (in kg) of the transfer vehicle:

$$m_{dry}^{tv} = B_t m_p^{tv} + C_t \tag{8}$$

Here, $B_t = 0.04545$, $C_t = 2279$ kg are the model coefficients for cryogenic vehicles; m_p^{tv} is the transfer vehicle propellant mass.

The wet mass of the transfer vehicle is defined as follows:

$$m_{wet}^{tv} = m_{dry}^{tv} + m_{p}^{tv}$$

HLS sizing equations. One observes that all mass estimating relationships defined in the previous subsection are linear with respect to the HLS mass components. Additional linear relations including the same mass components can be obtained by writing down the respective rocket equation expressions in the following form:

$$m_{fi} - E_i m_{0i} = 0, \quad E_i = \exp\left(-\frac{\Delta V_i}{I_{spi} g_0}\right) \tag{9}$$

Here, m_{0i} and m_{fi} are the total masses of the vehicle stack before and after the propulsive maneuver, respectively; ΔV_i is the delta-V of the maneuver; I_{spi} is the specific impulse of the vehicle performing the burn; g_0 is the Earth's standard gravitational acceleration. Combining the mass estimating relationships (1)-(8) with the rocket equation expressions (9) for the respective delta-V requirements yields a linear system with respect to the HLS mass components:

$$A\left(m_{dryPM}^{asc}\ m_p^{asc}\ m_{dry}^{dsc}\ m_p^{dsc}\ m_{dry}^{tv}\ m_{p1}^{tv}\ m_{p2}^{tv}\right)^T = B \tag{10}$$

In the mass vector above, m_{p1}^{tv} and m_{p2}^{tv} are the transfer vehicle propellant masses corresponding to ΔV_{tv1} and ΔV_{tv2} (see Table 1), respectively ($m_p^{tv} = m_{p1}^{tv} + m_{p2}^{tv}$).

The matrices A and B in Eq. (10) are defined as follows:

$$A = \begin{pmatrix} 1-E_a & -E_a & 0 & 0 & 0 & 0 & 0 \\ A_a - 1 & A_a + B_a & 0 & 0 & 0 & 0 & 0 \\ 1-E_d & 1-E_d & 1-E_d & -E_d & 0 & 0 & 0 \\ A_d & A_d & A_d - 1 & A_d + B_d & 0 & 0 & 0 \\ 1-E_{t1} & 1-E_{t1} & 1-E_{t1} & 1-E_{t1} & 1-E_{t1} & -E_{t1} & 1-E_{t1} \\ 0 & 0 & 0 & 0 & 1-E_{t2} & 0 & -E_{t2} \\ 0 & 0 & 0 & 0 & -1 & B_t & B_t \end{pmatrix}$$

$$B = \begin{pmatrix} -(1-E_a)m_{up} \\ -A_a m_{up} - C_a \\ -(1-E_d)m_{down} \\ -A_d m_{down} - C_d \\ -(1-E_{t1})m_{down} \\ 0 \\ -C_t \end{pmatrix},$$

$$E_a = \exp\left(-\frac{\Delta V_{asc1} + \Delta V_{asc2}}{I_{sp}^{asc} g_0}\right)$$

$$E_d = \exp\left(-\frac{\Delta V_{dsc}}{I_{sp}^{dsc} g_0}\right)$$

$$E_{t1} = \exp\left(-\frac{\Delta V_{tv1}}{I_{sp}^{tv} g_0}\right)$$

$$E_{t2} = \exp\left(-\frac{\Delta V_{tv2}}{I_{sp}^{tv} g_0}\right)$$

Parameters m_{up}, m_{down} in the equations above are defined as follows:

$$m_{down} = m_{cabin} + m_{consumables} + m_{PLD}^{down}$$
$$m_{up} = m_{cabin} + m_{consumables} + m_{PLD}^{up}$$

Solving system (10), one obtains necessary mass data to the size all three HLS elements.

Reference Mission. The following reference data was used for calculations. The number of crew supported by the HLS is 4 people. The payload mass delivered by the HLS to the lunar surface m_{PLD}^{down} is 500 kg; the payload mass returned from the lunar surface m_{PLD}^{up} is 250 kg. All vehicles are assumed to have a specific impulse I_{sp} of 450 s. The total crew support time $t_{support}$ and the number of extravehicular activity cycles n_{EVA} assumed for different Gateway orbits are given in Table 4.

Model validation. The 3-stage HLS parametric model described above produces the following masses of the HLS elements for the Gateway in the L2 4:1 NRHO (this orbit is close to the current NASA's reference Gateway orbit – L2 9:2 NRHO): the ascent vehicle wet mass is ~ 10700 kg, the descent vehicle wet mass is ~ 10700 kg, and the transfer vehicle wet mass is 7800 kg. In comparison, NASA's HLS preliminary requirements, as defined in the NextStep-2 Broad Agency Announcement Appendix E [7], stipulate that the wet masses at launch for those three elements shall not exceed 12000, 15000, and 15000 kg for the ascent, descent, and transfer vehicles, respectively (the Launch Vehicle considerations requirements). One observes that the model satisfies those requirements providing a comparable mass estimation for the ascent vehicle and somewhat lower estimations for the descent and transfer elements. Given that the NASA requirements represent the mass upper limits restricted by launch vehicle considerations and not vehicle mass estimations themselves as well as the fact that for a lighter ascent vehicle produced by the model, a lighter descent vehicle and an even lighter transfer vehicle would be needed, we assumed that the model produces plausible results and can be used for comparison purposes (i.e., can be used to compare landing systems corresponding to different Gateway orbits), even if the absolute values it produces prove to be somewhat off.

3. Results

The results of the HLS elements sizing for different Gateway orbits are presented in Table 4. The data in Table 4 accounts for the abort scenarios specified above (i.e., the ascent vehicle is sized based on $\Delta V_{asc2} = \Delta V_{asc2(ab)}$).

The sensitivity of the individual HLS element wet masses to the Gateway location are illustrated in Fig. 1-3. The sensitivity of the HLS total wet mass to the Gateway location is illustrated in Fig. 4. Gateway orbits are arranged along the *x*-axis by their energy – from orbits with lower energy (those are closer to the Moon) on the left to orbits with higher energy on the right (those are farther from the Moon). In order to understand the impact of the abort operations on HLS sizing, we used two sizing strategies: within the first, baseline, strategy, the delta-V data that accounted for the abort scenarios ($\Delta V_{asc2} = \Delta V_{asc2(ab)}$) was used to plot the graphs (solid lines in Fig. 1-4); within the second strategy, the HLS vehicles were sized solely based on the nominal operations delta-Vs ($\Delta V_{asc2} = \Delta V_{asc2(nom)}$; dotted lines in Fig. 1-4). We chose the HLS sized for the L2 4:1 NRHO (which is close to the current NASA's reference Gateway orbit – L2 9:2 NRHO) as a baseline for comparing systems sized for different Gateway orbits. In each figure, the percent change in the respective wet mass compared to the baseline HLS is indicated next to the respective point of the graph.

As one can see from the figures, the Gateway location affects all three of the HLS elements with the transfer vehicle being the most affected. It is not surprising as the transfer vehicle is directly dependent on the delta-Vs for the transfers between the Gateway and the 100-km LLO which change with the Gateway location. The ascent vehicle also involves a pathway between the LLO and the Gateway, so its delta-V requirements also differ for different Gateway orbits which affects the element's mass. Despite the fact that the delta-V for the descent vehicle is basically constant, its mass also changes due to the changes in the ascent vehicle's mass which it has to land.

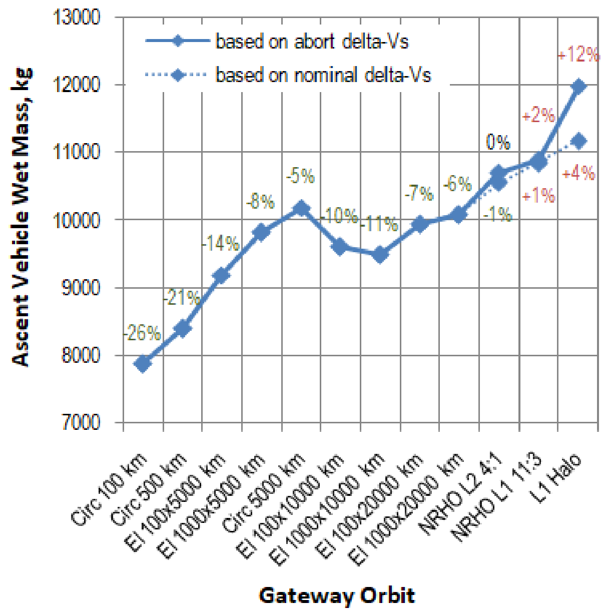

Fig. 1. Ascent vehicle wet mass sensitivity to the Gateway location

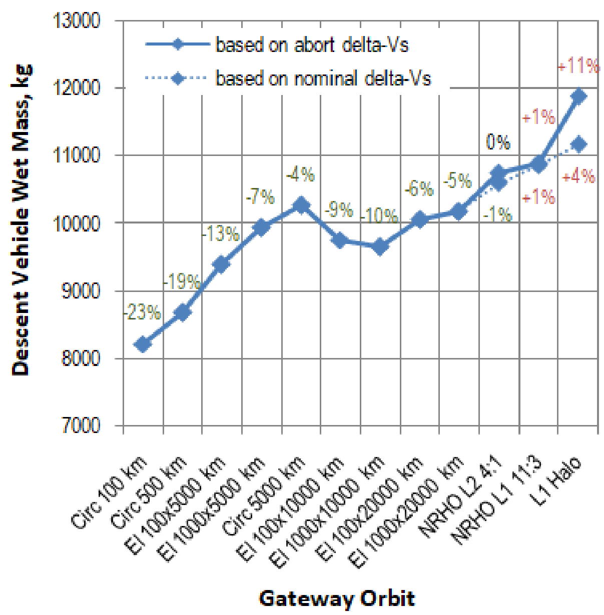

Fig. 2. Descent vehicle wet mass sensitivity to the Gateway location

As the sensitivity graphs show, among the Gateway orbits considered, the NRHOs and the halo orbit impose the highest requirements on the human landing system and each of its three stages. Not only the nominal transfers between the Gateway and LLO in this case are more expensive (in terms of delta-V) but the abort operations consideration makes this delta-V price even higher compared to simpler cases of polar circular and elliptic orbits (for which abort operations are not that different from the nominal ones). The abort operations effect is the most prominent for the halo orbit where it results in a 6% increase in the HLS total wet mass; in case of both NRHOs, for the selected Gateway-to-LLO/ LLO-to-Gateway transfer time of 0.5 day, the respective increase in mass is not that significant and constitute only about 1% of the HLS total wet mass. It is worth noting, however, that NRHOs are characterized by the most complex abort operations among all Gateway orbits – unlike polar circular, polar elliptic, and L1 halo orbits, for which the abort operations are practically the same as the nominal ones, NRHOs require using different abort strategies depending on the abort day to minimize the abort delta-Vs [8].

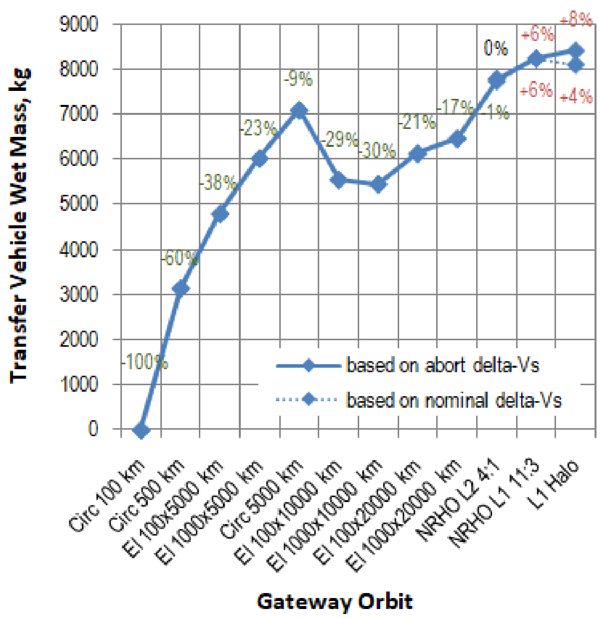

Fig. 3. Transfer vehicle wet mass sensitivity to the Gateway location

The lightest HLS corresponds to the Gateway in the 100-km polar circular orbit. In this case, the transfer vehicle is not needed at all (hence its zero mass), and the ascent and descent vehicles are 2658 kg (~ 26%) and 2372 (~23%) lighter than their baseline NRHO versions; the total HLS mass for the 100-km orbit is 45% lighter than the one for the 4:1 NRHO. The heaviest HLS corresponds to the Gateway in L1 halo orbit; its wet mass is 10% higher than the one of the baseline HLS.

Analyzing Table 2, one also observes that polar circular and elliptic orbits have no serious limitations on the surface access from the Gateway station as well as on the surface stay duration – transfers to the lunar surface in this case are possible every or, at most, every other day, and the surface stay can be practically of any length (i.e., any number of days). The emergency returns to the station are the shortest in this case and last ~ 2 days (for the highest orbits) or less. The situation is more complicated in case of NRHOs and the L1 halo orbit. The surface access for those orbits is restricted to 1-2 times per month, and the optimal stay duration is a 7.4-, 8-, or 12-day multiple for the L2 NRHO, L1 NRHO, and L1 halo, respectively. The emergency returns are also longer and last up to 4-5 days for NRHOs, and up to 4-8 days for L1 halo. It is worth noting that, in case of NRHOs, the restriction of the surface access to 1-2 times per month is due to the necessity of having an abort opportunity in case of emergency in LLAO. If this restriction is lifted, transfers to the surface from the Gateway in NRHO are possible every 7-8 days.

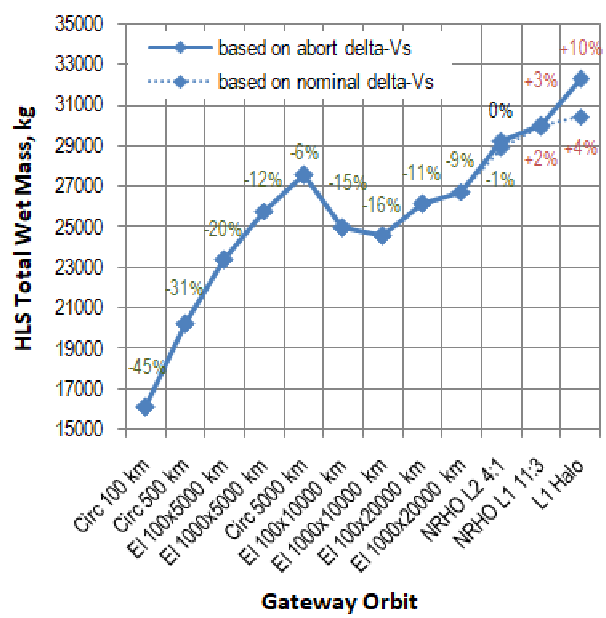

Fig. 4. HLS total wet mass sensitivity to the Gateway location

4. Discussion

Our findings suggest that, if the HLS performance were the only factor defining the choice of the Gateway location, then the lowest possible orbit – a 100-km circular polar orbit – would be the optimal solution providing the lowest HLS mass* and the simplest HLS

* One might argue that, due to the closeness of the higher Gateway orbits to the Earth (and the lower delta-Vs of the respective LEO-to-Gateway transfers), when scaled back to LEO, the initial mass of the HLS-related stack in LEO might be lower for the higher Gateway orbits, indicating lower launch costs. However, as it is shown in [8], it is not the case. The increase in the HLS wet mass for higher Gateway orbits due to their being farther from the Moon outweighs the benefit of those orbits having lower LEO-to-Gateway delta-V costs due to their being closer to the Earth, and the initial mass in LEO of the HLS-related stack grows as well in this case.

operations. There are, however, two important remarks that outline the limitations of this conclusion application and identify potential areas of future research into the impacts of the Gateway location choice on Gateway-involving projects.

First, HLS performance is not the only factor defining the choice of the Gateway location. The most significant of the other factors is access to the Gateway from Earth by a crew vehicle. Currently, the Orion Crew Vehicle is planned to be used for that purpose. According to [11], the Orion propulsion system provides the total delta-V of 1340 m/s which makes the lowest two orbits (100-km and 500-km) inaccessible for this particular spacecraft (see Table 5).

The current NASA's choice for the Gateway orbit is an L2 9:2 NRHO (which is close in its performance to the L2 4:1 NRHO analyzed here) which is reachable for the Orion but is among the costliest for the HLS. A quick look at Table 4 shows that a lower elliptic orbit might be a less expensive alternative to the NASA's choice. The data in the table, however, provides only minimum delta-V requirements which do not account for potential emergency returns of the crew vehicle from the Gateway to the Earth. These emergency operations for circular and elliptic Gateway orbits might require expensive plane-changing maneuvers and make them somewhat less attractive than they appear in the table. So, a logical next step in identifying the optimal Gateway orbit would be analyzing the impact of the Gateway location on the crew vehicle operations with the goal of identifying the lowest Gateway orbits which would still be reachable by the current Orion.

Some other factors to consider when choosing the Gateway location include station-keeping costs, shadow avoidance, communication with Earth and the lunar landing site, and extensibility to future Mars missions [2]. Another relevant factor is impact of the Gateway orbit on potential future lunar propellant transportation architecture (for future projects involving propellant production on the lunar surface). If the Gateway is considered part of future Mars exploration and lunar propellant production campaigns, then the choice of its orbit might lock down some key architectural decisions for those initiatives. Hence, analysis of this kind of impact seems to be another important line of future investigation into how the Gateway orbit choice affects Gateway-related projects.

Second, the results stated here are fair for the adopted 'minimalist', single-landing-site, surface access strategy. This strategy produces the least delta-V costs for regular access to the lunar South Pole (and the lunar North Pole) for most of the Gateway orbits considered. Adding the requirement for the HLS to be able to visit other lunar sites will increase the respective HLS delta-V requirements (other lunar sites either require more expensive surface abort operations (for circular and elliptic Gateway orbits) or are generally more expensive to reach (for NRHOs as Gateway orbits)). Due to the specifics of the respective flight geometries, the costs of visiting other sites will be different for different Gateway locations with lower Gateway orbits likelier to have higher costs. As a result, within another surface access strategy, the conclusions regarding the impact of the Gateway location on the HLS performance might differ from the ones presented here. Hence, another line of further investigation might include studying impact of the Gateway location on the costs of visiting lunar sites other than the lunar poles as well as developing surface access strategies that would still satisfy potential stakeholders' needs to visit other sites but at a lower cost than the 'maximalist' global-access strategy which implies regular access to any location on the lunar surface by the same HLS under the same conditions.

5. Conclusions

This paper explored the impact of the Gateway location on the HLS sizing in case of having a permanent habitable base at the lunar South Pole. A 3-stage LOX/LH2 HLS architecture was assumed and 12 different Gateway orbits were considered including 3 polar circular and 6 polar elliptic lunar orbits, 2 NRHOs and 1 L1 halo orbit.

Our analysis has revealed that all three HLS elements (ascent, descent, and transfer vehicles) are affected by the Gateway location, and the HLS mass characteristics rise rather quickly with the increase in the Gateway orbit energy. The lightest HLS corresponds to the Gateway in a 100-km polar lunar orbit, the heaviest to the L1 halo orbit. NRHOs, which are NASA's current baseline option for the Gateway location, are among the orbits with heavier HLS. The total wet mass of the lightest HLS in a 100-km orbit is 45% lower than the total HLS wet mass for an L2 4:1 NRHO (which is close to the current NASA's reference Gateway orbit, an L2 9:2 NRHO).

Circular and elliptic polar lunar orbits as Gateway orbits are characterised by lower transfer times, the simplest abort operations and the absence of any serious limitations on the surface access frequency and stay times. The L1 halo orbit is characterised by the longest transfer times, relatively simple but expensive (in terms of delta-V) abort operations. NRHOs as Gateway orbits are characterized by complicated abort operations. Both NRHOs and L1 halo have certain limitations on the surface access frequency and stay times.

This increase in the HLS mass and operations complexity is a drawback to the benefits that a Gateway in NRHO can potentially provide.

Additional investigations that would either justify this NASA's Gateway orbit choice or find a better alternative include analyzing impact of the Gateway location on other Gateway-involved transportation systems such as the crew vehicle to transport people between the Earth and the Gateway, potential future lunar propellant transportation architecture, and potential Mars exploration campaign.

The results presented here are fair for the ‘minimalist’, single-landing-site, surface access strategy (with the landing site at one of the lunar poles). So, another line of further investigation might include studying impact of the Gateway location on the costs of visiting lunar sites other than the lunar poles.

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Table 1. Delta-V requirements for the individual HLS elements for different Gateway orbits

Gateway orbit	**Ascent Vehicle**				**Descent Vehicle**		**Transfer Vehicle**	
	In-plane ascent	Nominal LLDO-to-GW transfer	Abort LLAbO-to-GW transfer	Abort LLAO-to-GW transfer*	In-plane descent	Abort LLAO-to-GW transfer*	Nominal GW-to-LLAO transfer	Nominal LLAO-to-GW transfer
	ΔV_{asc1}	$\Delta V_{asc2(nom)}$	$\Delta V_{asc2(ab)}$	-	ΔV_{dsc}	-	ΔV_{tv1}	ΔV_{tv2}
Circular 100 km	1900	0	0	0	1900	0	0	0
Circular 500 km	1900	160	160	160	1900	160	160	160
Circular 5000 km	1900	710	710	710	1900	710	710	710
Elliptic 100x5000 km	1900	420	420	420	1900	420	420	420
Elliptic 100x10000 km	1900	520	520	520	1900	520	520	520
Elliptic 100x20000 km	1900	590	590	590	1900	590	590	590
Elliptic 1000x5000 km	1900	510	510	510	1900	510	510	510
Elliptic 1000x10000 km	1900	580	580	580	1900	580	580	580
Elliptic 1000x20000 km	1900	630	630	630	1900	630	630	630
NRHO L2 4:1	1900	780	820	740	1900	740	780	740
NRHO L1 11:3	1900	830	840	760	1900	760	830	760
L1 Halo Az = 12000 km	1900	790	980	950	1900	950	790	790

* Abort from LLAO is assumed to be implemented either by the ascent or descent vehicle, depending on the nature of the emergency. Since each of the vehicles have enough nominal delta-V onboard to perform the abort transfer, the delta-Vs in this column do not affect HLS sizing.

Table 2. Transfer times for nominal and abort operations for different Gateway orbits

Gateway orbit	**Nominal GW-LLAO/ LLDO-GW transfer**	**Nominal LLAO-GW (transfer vehicle)**	**Abort from LLAO**	**Abort from surface**	**Surface access frequency**	**Surface departure frequency**	**Optimal surface stay time**
Circular 100 km	-	-	-	4 hr	every 2 hrs	every 2 hrs	any
Circular 500 km	1.2 hr	8 hr	8 hr	6 hr	every 3 hrs	every 3 hrs	any
Circular 5000 km	3.5 hr	7 hr	7 hr	8 hr	every 14 hrs	every 14 hrs	any
Elliptic 100x5000 km	3.5 hr	10.5 hr	10.5 hr	10.5 hr	every 7 hrs	every 7 hrs	any
Elliptic 100x10000 km	7 hr	21 hr	21 hr	21 hr	every 14 hrs	every 14 hrs	any
Elliptic 100x20000 km	16 hr	48 hr	48 hr	48 hr	every 32 hrs	every 32 hrs	any
Elliptic 1000x5000 km	4 hr	12 hr	12 hr	12 hr	every 8 hrs	every 8 hrs	any
Elliptic 1000x10000 km	8 hr	23 hr	23 hr	23 hr	every 16 hrs	every 16 hrs	any
Elliptic 1000x20000 km	17 hr	51 hr	51 hr	51 hr	every 34 hrs	every 34 hrs	any
NRHO L2 4:1	12 hr	up to 4 day	up to 4 day	up to 5 day	1-2 times per month*	every 7.4 days	7.4-day multiple
NRHO L1 11:3	12 hr	up to 5 day	up to 5 day	up to 5 day	once a month*	every 8 days	8-day multiple
L1 Halo Az = 12000 km	up to 3.6 day	up to 10.4 day	up to 7.6 day	up to 3.6 day	every 12 days	every 12 days	12-day multiple

* Limited by LLAO abort opportunities (based on preliminary findings from [8])

Table 3. HLS reference mission data

	Surface stay, day	$t_{support}$, day	n_{EVA}
Circular 100 km	7	7	7
Circular 500 km	7	8	7
Circular 5000 km	7	8	7
Elliptic 100x5000 km	7	8	7
Elliptic 100x10000 km	7	9	7
Elliptic 100x20000 km	7	10	7
Elliptic 1000x5000 km	7	8	7
Elliptic 1000x10000 km	7	9	7
Elliptic 1000x20000 km	7	10	7
NRHO L2 4:1	7.4	9	7
NRHO L1 11:3	8	10	8
L1 Halo Az = 12000 km	7	15	7

Table 4. HLS elements sizing

Gateway Orbit	**Ascent Element**			**Descent Element**			**Transfer Vehicle**		
	m_{dry}^{asc}, kg	m_{p}^{asc}, kg	m_{wet}^{asc}, kg	m_{dry}^{dsc}, kg	m_{p}^{dsc}, kg	m_{wet}^{dsc}, kg	m_{dry}^{tv}, kg	m_{p}^{tv}, kg	m_{wet}^{tv}, kg
Circular 100 km	5042	2846	7889	2413	5810	8223	-	-	-
Circular 500 km	5178	3228	8406	2533	6152	8685	2316	823	3140
Circular 5000 km	5529	4659	10187	2943	7332	10275	2488	4602	7090
Elliptic 100x5000 km	5333	3860	9193	2714	6673	9387	2388	2405	4793
Elliptic 100x10000 km	5449	4161	9611	2810	6950	9760	2421	3127	5548
Elliptic 100x20000 km	5549	4394	9944	2887	7170	10057	2446	3681	6127
Elliptic 1000x5000 km	5391	4096	9487	2782	6868	9650	2417	3033	5450
Elliptic 1000x10000 km	5490	4327	9817	2858	7086	9944	2442	3579	6020
Elliptic 1000x20000 km	5577	4509	10086	2920	7264	10184	2461	3999	6460
NRHO L2 4:1	**5664**	**5037**	**10701**	**3062**	**7672**	**10733**	**2519**	**5274**	**7793**
NRHO L1 11:3	5733	5147	10880	3103	7790	10893	2538	5708	8246
L1 Halo	6122	5863	11985	3358	8521	11879	2546	5870	8416

Table 5. Crew vehicle minimum delta-V requirements

Gateway Orbit	**Round-trip LEO-Gateway-Earth minimum delta-V*, m/s**
Circular 100 km	1640
Circular 500 km	1560
Elliptic 100x5000 km	840
Elliptic 1000x5000 km	980
Circular 5000 km	1220
Elliptic 100x10000 km	640
Elliptic 1000x10000 km	760
Elliptic 100x20000 km	500
Elliptic 1000x20000 km	600
NRHO L2 4:1	860
NRHO L1 11:3	740
L1 Halo Az = 12000 km	1260

* The minimum requirements assume that the crew vehicle arrives at and departs the station at optimal dates (the trans-lunar injection is performed by a human-rated Earth departure stage; the crew vehicle performs the Gateway orbit insertion and departure maneuvers). Additional requirements for potential abort Earth return opportunities due to an emergency at the station might result in larger delta-Vs than those in the table. The respective delta-V increases are specific to the Gateway location.