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REFINED MISSION ANALYSIS FOR HERACLES – A ROBOTIC LUNAR SURFACE SAMPLE RETURN MISSION UTILIZING HUMAN INFRASTRUCTURE

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In the frame of the International Space Exploration Coordination Working Group the European Space Agency is participating in the planning of future exploration architectures. The mission concept for the robotic lander mission (Human Enhanced Robotic Architecture and Capability for Lunar Exploration and Science – HERACLES) has matured meanwhile. The mission aims for a human assisted sample return from the lunar surface, while at the same time providing a qualification opportunity for technologies required for a crewed lunar lander. Human spaceflight rating is required for parts of the mission, since the sample return shall not be via a direct return trajectory, but the samples shall be transported via Orion, and thus docking of the robotic lander to the LOP-G will be required. This paper shall provide an update on the current mission design as agreed between the international partners and the associated mission analysis as all the intermediate and final orbits have been selected for the baseline. The implications of the design decision as well as some alternatives that can serve as a backup scenario will be presented as well. The paper will first present the baseline mission sequence and will then focus on aspects of particular interest as e.g. the strong limitation in the launch window design and the rendezvous and docking strategy.

INTRODUCTION

HERACLES is intended as a human-robotic architecture in the frame of preparations for human exploration activities on the lunar surface. On international level, the exploration goals and objectives are coordinated by the ISECG. One product of the ISECG work is the Global Exploration Roadmap (GER) [1], which summarizes the goals and objectives and derives a set of mission themes.

The roadmap represents a sequence of missions based on the current understanding of the partners' policies and plans, leading to human exploration of Mars and the Moon in a step-wise manner. A graphical overview of the roadmap is given in Figure 1.

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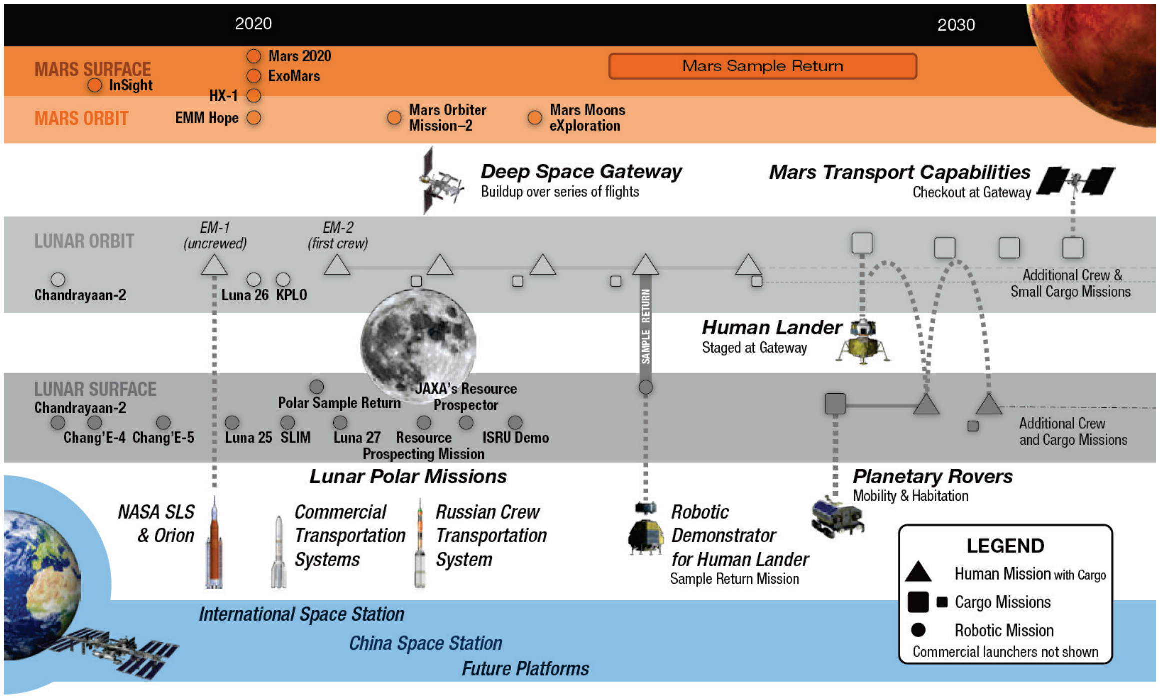

Figure 1: Global Exploration Roadmap of the ISECG, the framework for the HERACLES mission, depicted as Robotic Demonstrator for Human Lander in the Roadmap

The top-level objectives of the GER are to:

1. Develop exploration technologies and capabilities
2. Engage the public in exploration
3. Enhance Earth safety
4. Extend human presence
5. Perform science to enable human exploration
6. Perform space, Earth, and applied science
7. Search for life
8. Stimulate economic expansion

The second iteration of the GER published in August 2013 noted that new mission concepts, such as human-assisted sample return and tele-presence should be further explored, increasing understanding of the important role of humans in exploration for achieving common goals. Responding to this observation, ESA has initiated in 2015 the HERACLES study process for understanding the role human-robotic partnership for lunar exploration, defining concepts, and assessing the programmatic feasibility. ESA invited all ISECG participating space agencies to engage in the study. The study is fully aligned with the ESA Space Exploration Strategy and the mission statement is formulated as follows:

Mission Statement

The Human-Enhanced Robotic Architecture and Capability for Lunar Exploration and Science (HERACLESGT) is to establish key elements and capabilities for sustainable human explo-

ration of the Moon and human-robotic exploration of Mars by implementing lunar surface operations while maximising opportunities for unprecedented scientific knowledge gain

The paper will in the following present the advanced mission analysis activities conducted for HERACLES. For this it is important to understand the baseline mission scenario as depicted in Figure 2. The depicted scenario shows the initial delivery of elements and it could in the future be extended with reusable elements.

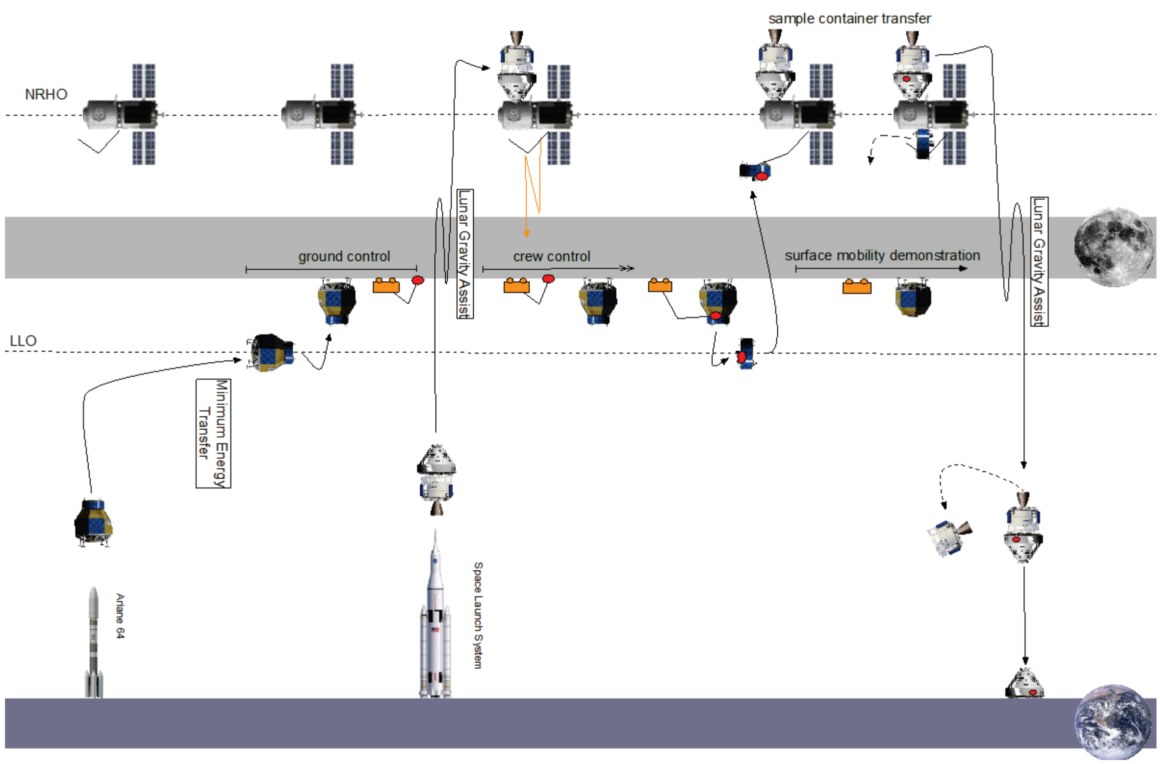

Figure 2: HERACLES baseline mission scenario

As a baseline, the first HERACLES mission starts with the launch of a mid-sized launch vehicle (e.g. Ariane 64) with a direct injection into the trans-lunar orbit. As an alternative the launch can be in GTO.

At Earth departure the HERACLES lunar module comprising the Lunar Descent Element (LDE), Lunar Ascent Element (LAE) and Surface Mobility Element is either directly injected into the trans-lunar trajectory or performs a manoeuvre sequence to achieve trans-lunar injection (TLI) for a transfer to the Moon. The LDE main engine performs the large injection manoeuvres.

As a transfer options the Weak Stability Boundary (WSB) or minimum energy transfers were studied, however, at the time being the utilization of a novel cryogenic main engine on the LDE prohibits the utilization of the WSB transfer due to the long time between Earth departure and lunar orbit insertion (LOI). During the transfer phase, the LDE must minimise cryogenic propellant boil-off.

Navigation operations occur during the transfer to enable LOI with the required precision. Further tracking is performed after the initial LOI and OCMs could possibly be flown for a precise LLO injection, however, at it will be shown later in the paper this is considered unnecessary.

On LLO, the powered descent is preceded by a periselenium lowering manoeuvre and more tracking to initialise the LDE GNC for descent. The descent proper is initiated by ignition of the LDE main engine. It is assumed that the landing is to occur during daylight conditions at the landing site with TBD illumination constraints – a landing in darkness is not required. During descent the LDE controls the vehicle attitude to follow the descent profile to a high gate arrival (HGA) above the lunar landing site. HGA is the interface point, at which precision requirements of the descent are formulated. The part of the mission after HGA is referred to as final descent and comprises of the reduction of remaining velocity and altitude while avoiding collision with terrain. The final approach ends in the hovering phase, the goal of which is to precisely acquiring the reference altitude (TBD), to steer clear of any inacceptable terrain (rocks, craters or steep slopes), to zero out any horizontal motion (within the specifications of the control system), and to perform main engine cut-off (MECO) for final ballistic descent to the surface.

On the surface, the lunar module is commissioned for surface mode. Then, the LDE extends the mechanism for rover deployment. The rover egresses the LDE and starts the surface campaign. At this stage the LDE can be decommissioned and deactivated. Initial exploration of the surface by the rover is supported by ground control and time-tagged commanding until crew arrives on the LOP-G.

The crew transfer is achieved by the launch of the crew vehicle into Lunar Transfer Orbit (LTO, outbound leg). The transfer scenario for the crew vehicle is a lunar gravity-assist transfer towards the LOP-G, which we assume to be stationed on a Near-Rectilinear Halo Orbit (NRHO). Once the crew is present, the crew-supported surface mobility operations can start. The rover then is commanded to perform sample collection and transfer the sample container to the LAE. At the end of the sample collection phase, which can take multiple lunar day-night cycles (current assumption is 70 days, actual crew mission duration is the more limiting factor), the LAE subsystems are commissioned and tested for ascent. On the subsequent inbound leg of the mission, the LAE ascents into an initial orbit with the aposelenium at the altitude of the intermediate circular LLO and the periselenium high enough such that the initial orbit will not lead to an impact on the surface for at least two weeks if no manoeuvres are performed after the insertion. From the initial orbit, the intermediate LLO is achieved by a circularisation manoeuvre after an orbit determination campaign. After tracking and navigation the LAE initiates the transfer to the LOP-G, which is followed by the rendezvous and berthing.

MISSION ANALAYSIS ASPECTS

The following mission analysis aspects will be treated in greater detail:

1. Transfer Window Constraints and Design
2. Lunar Orbit Injection
3. Rendezvous and Docking
4. Disposal of mission elements

Transfer Window Constraints and Design

Designing the transfer window for HERACLES turns out to be difficult due to three constraints/assumptions that were stated for the S/C stack:

1. Launch from Kourou on an Ariane 64 launcher with maximum payload performance

2. Limitation of the transfer time to 20 days due to the utilization of cryogenic propellant
3. Landing of the LDE in the lunar morning or at a specific local time

The first assumption of a launch utilizing the Ariane 64 launch vehicle places a strong limitation on the available launch days. The requirement of having a maximum performance launch for maximum payload utilization of the launcher results in a low inclination of the injection orbit near 5 Deg. Since this inclination is significantly lower than the Moon's orbit about the Earth transfer opportunities will exist only twice per month when the declination of the moon with respect to the Earth is low enough.

In theory the inclination of the lunar arrival orbit can now be selected such that the vehicle will pass over the landing site for a co-planar descent from the arrival orbit. However, the landing has to occur at a specific local time. This might be e.g. either the lunar morning, to allow for a full lunar day of surface operations or during lunar noon to get the best illumination of a high latitude landing site in case the LDM will rely on optical sensors for the landing. This illumination constraint is not automatically fulfilled and it is actually fulfilled only one per month.

To achieve the optimal illumination required for the landing the inclination of the arrival orbit can be chosen such that the landing site will pass underneath the orbit at the correct local time, but in the worst case a loitering time of up to one month will be required. Such a long transfer time is currently not acceptable due to the utilization of cryogenic propellants on the LDM. Thus, a lot of transfer opportunities from the Earth towards the moon must be skipped due to a too long transfer duration. So an important question is which minimum mission duration is required to allow for a transfer between Earth and the landing site at least once a month? This can be calculated by considering the above constraints on the Earth departure orbit as well as the illumination constraints.

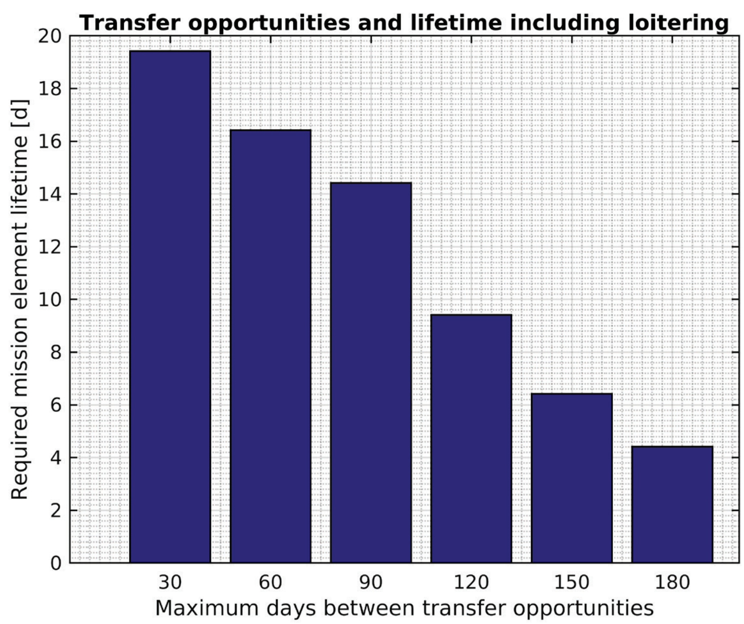

Figure 3: Days between transfer opportunities for different acceptable mission durations

A too long time between two consecutive transfer opportunities is undesirable for several reasons. The launcher manifest of the launch service provider might be severely interrupted if a transfer opportunity cannot take place and the launch pad as well as the preparation facilities are blocked for several months. In addition the HERACLES mission is in the optimal case phased together with a crewed mission to the LOP-G and thus a too long launch delay would be unacceptable. To allow for a transfer opportunity at least once per month a transfer duration of 20 days should be acceptable for the LDM at it can be seen from Figure 3.

Lunar Orbit Insertion

The lunar orbit insertion maneuver ideally injects the S/C into the desired 100 km circular LLO in a single injection burn. But an important factor when performing the injection burn is the manoeuvre execution error. The injection burn is usually undershoot to avoid a too low periselenium altitude in case of an over-performance of the manoeuvre, e.g. for a 3% over-performance on a typical LOI the resulting periselenium altitude would already be below the lunar surface and the S/C stack would crash.

Typically a series of correction burns is performed after injection into an elliptic LLO to establish the baselined circular 100x100 km LLO. However, in case of HERACLES the number of main engine firing shall be limited due to the associated cryogenic propellant losses per firing. It was thus investigated if the descent to the surface location can be executed from an uncorrected and thus elliptical LLO and if a significant penalty in ΔV would arise. This requires that the periselenium lowering burn can take place at any true anomaly and altitudes larger than 100 km. The resulting state prior to the powered descent thus shows a larger variation in velocity when compared to the 100 km LLO case and the GNC algorithm for the powered descent phase must take these larger variation into account, although the orbit determination prior to the descent will provide the algorithm with the expected state of the uncorrected LLO. When assuming the 3% manoeuvre execution error a 100 x 343 km orbit might result from the required undershooting of the orbit. Surprisingly, when calculating the total descent DeltaV the penality is only 1.8 m/s, so the baseline decision for HERACLES is to not correct the achieved LLO.

In addition it can be expected that the manoeuvre execution error will be by far smaller than 3% due to the hardware carried for the powered descent phase. An uncorrected LLO will result in a slight timing error for passing over the landing site and thus a slight cross-track component would need to be corrected. A further aspect to be considered in the mission timeline is that the time of landing could vary by up to one orbital period, but this is traded against the missing complexity of additional correction burns.

Surface Operations

The surface operations is not part of the mission analysis, however, mission analysis products are required for a proper design. Mission analysis mainly provides visibility and illumination information with respect to e.g. the Earth, the Sun or the LOP-G or any other S/C that could be used for data relay. Dedicated software tools were developed to support these activities and are described in a previous paper [4].

Ascent from surface to LLO

After surface science operations, the lander shall ascend and rendezvous with the LOP-G such that the collected samples can safely be transferred to the Orion vehicle for Earth return. First, it performs a powered ascent to a parking 100x100km LLO, from which it departs. To achieve proper phasing with the LOP-G and for a ΔV optimal ascent a loitering period in LLO might be required.

The inclination of the LLO is mainly constrained by the landing site. If a ΔV optimal coplanar ascent shall be achieved the latitude of the landing side dictates the minimum achievable inclination. As limit case, a polar site allows to reach only 90 deg inclination LLOs (excluding expensive a-posteriori plane change maneuvers). Note that the LLO inclination, if referred to the lunar equator, is defined in a Moon-centred equatorial frame; on the contrary, NRHOs and transfer trajectories are defined in the Earth-Moon co-rotating frame. Thus, the transfer epoch affects the relationship between the two frames, and the relative inclination can affect the Delta-V of the transfer from LLO to NRHO, discussed in detail in the following Section.

If the inclination of the LLO is not fixed, one can find an optimal inclination at each epoch, in order to reduce the overall Delta-V. Otherwise, the transfer cost may have variations up to 100 m/s more of the nominal value, if the LLO inclination is constrained.

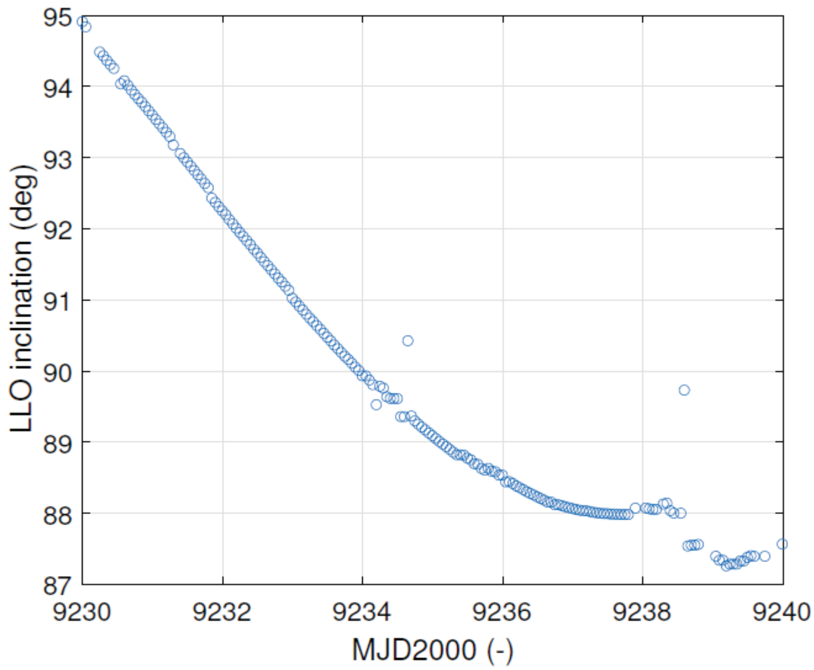

Figure 4: Optimal LLO inclination for transfer to NRHO

As operative example, Figure 4 portrays the optimal LLO inclination for different transfer epochs in 2025. The optimal LLO inclination is achieved in the rendezvous optimization. It is, in fact, the inclination which minimizes the Delta-V to transfer to NRHO, corresponding to the first maneuver in Figure 6. The relative orientation of the lunar pole in the Earth-Moon rotating frame is thus responsible for the different optimal geometry, according to the transfer epoch.

Rendezvous with LOP-G

Special care must be taken in the design of the rendezvous scenario with the LOP-G for two main reasons:

1. The correction manoeuvres after departure from the LLO are time critical and the correction DeltaV grows significantly with time
2. The rendezvous is with a potentially crewed and human rated vehicle and thus special precautions should be applied

The first point is dealing with the LLO departure burn execution error. Since the energy of the vehicle will be raised to almost escape velocity the manoeuvre to leave the LLO for rendezvous on the NRHO has a size of about 600-700 m/s. With a worst case assumption of a 3% manoeuvre execution error the resulting periselenium velocity will have an error of up to 21 m/s and the resulting departure trajectories can significantly deviate from the one intended as it can be seen in Figure 4.

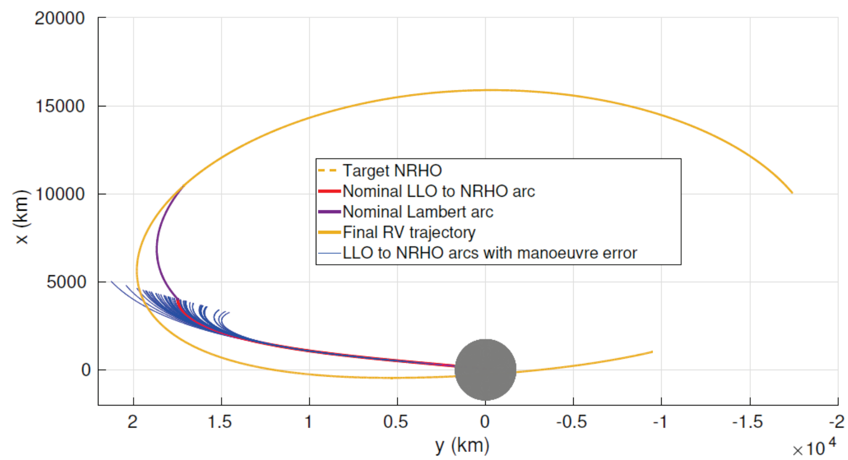

Figure 5: Rendezvous trajectory towards NRHO with manoeuvre execution error at periselenium. The perturbed trajectories are plotted for 1 day after the periselenium manoeuvre

The 99% percentile of the transfer correction manouevre is provided in Table 1. The size of the manoeuvre can be significant, requiring an early correction and a change in the operational concept, since e.g. the typical Sun-Earth Libration Point mission allows for a correction up to 48 hours into the mission. A second line of action is to reduce the manoeuvre execution error, which in a first assumption scales linearly with the correction DeltaV.

Table 1: Size of the TCM for different times of the correction maneuver, assuming 3% magnitude error

	TCM performed		
	After 12 hours	After 24 hours	After 48 hours
TCM (m/s)	105	156	272

As initial solution to the problem, the rendezvous design was based on the use of intermediate trajectories, reaching first an elliptical lunar orbit, then an intermediate phasing NRHO, and eventually performing the terminal maneuver, to reach the LOP-G in the target NRHO [3]. This approach was employed to reduce the stochastic component of the Delta-V budget; the intermediate phasing orbits are employed to mitigate the effect of the LLO burn dispersion, and are able to relax the phasing and timing constraints.

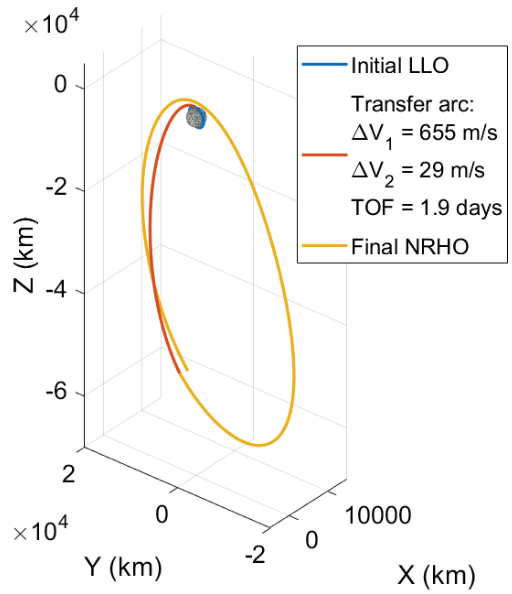

Figure 6: Trajectory from LLO to NRHO

As second iteration in the design process, it was observed that the propellant saved by a more accurate execution of the manoeuvre can easily compensate for additional hardware used to increase the accuracy of the manoeuvre execution. Thus, a more straightforward rendezvous strategy can be employed, where a direct LLO-NRHO transfer arc is designed. If the accuracy of the maneuver is increased, the stochastic TCM Delta-V budget is significantly reduced, and such direct transfer may be flown without significant increase in cost. Table 2 reports the TCM Delta-V, if the maneuver at LLO possesses a 0.3% execution error. It is noted how the accuracy increase is proportional to the Delta-V savings, thus suggesting that such strategy is beneficial for the mission. The direct transfer option, furthermore, requires less maneuvers and a more straightforward operational timeline.

Table 2: Size of the TCM for different times of the correction maneuver, assuming 0.3% magnitude error

	TCM performed		
	After 12 hours	After 24 hours	After 48 hours
TCM (m/s)	12	19	31

The rendezvous trajectory from LLO to NRHO is depicted in Figure 5. For an unmanned ascent element, no strict timing constraints were assumed, and the transfer duration may last up to 4 days, according to the initial LLO orientation, LLO loitering time and phasing with the LOP-G. It is remarked how, in case of stringent requirements on the time of flight, e.g. if a human vehicle is involved, the transfer arc may be shortened up to 12 hours, at the expense of a Delta-V increase of about 50 m/s. The values reported are related to a sample scenario, computed in a full ephemeris model, and might slightly vary according to the epoch of the transfer.

Disposal of Mission Elements

The disposal of mission elements might not be the biggest concern of the architecture design today, but to adhere with space debris mitigation requirements a look at the end-of-life trajectories will be required sooner or later in the mission design. Another aspect is the safe distance the LAE should keep from the LOP-G after undocking. A collision must be avoided by all means and thus a ΔV must be allocated anyway. Naturally the LDE as well as the rover will remain on the surface of the moon. However, the LAE has finished its mission after un-docking from the LOP-G. Due to the nature of this orbit several possibilities do exist:

- The LAE can crash on the moon
- The LAE can return towards the Earth and re-enter via a reverse WSB transfer
- The LAE can perform an heliocentric escape

By choosing the appropriate epoch for the undocking and a small departure manoeuvre from the NRHO these three possibilities can be triggered. With a Delta-V between 2-5 m/s, all the disposal possibilities can be achieved:

- A periselene maneuver may be used to crash on the Moon, through reduction of the orbit energy. The time to reach the surface, and the crash site location, depend on the maneuver magnitude; the epoch plays a minor role, dictating the exact surfacelongitude As depicted in Figure 7, a disposal maneuver at periselene, in anti-velocity direction, will lead to a crash in the northern lunar emisphere. The corresponding time to collision is reported in Figure 8.
- An aposelene maneuver triggers NRHO escape. The maneuver epoch and direction play a fundamental role in dictating the subsequent trajectory shape, and shall be designed with care if a heliocentric escape is targeted.

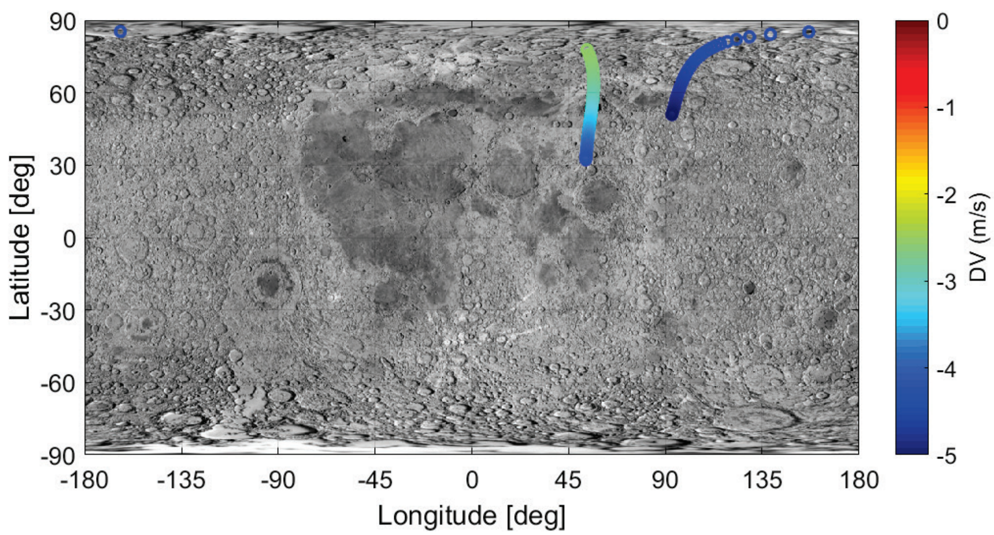

Figure 7: Sample crash locations on the lunar surface, after disposal from NRHO

Currently a return to Earth with subsequent re-entry into the atmosphere does not seem to be an attractive options due to the increased operational cost. To achieve a controlled re-entry the LAE must be operated during the long transfer phase via the WSB, orbit determination must take place and the LAE must execute transfer correction maneuvers. Since there is not advantage in terms of disposal this option has not been further investigated due to the increased operational cost.

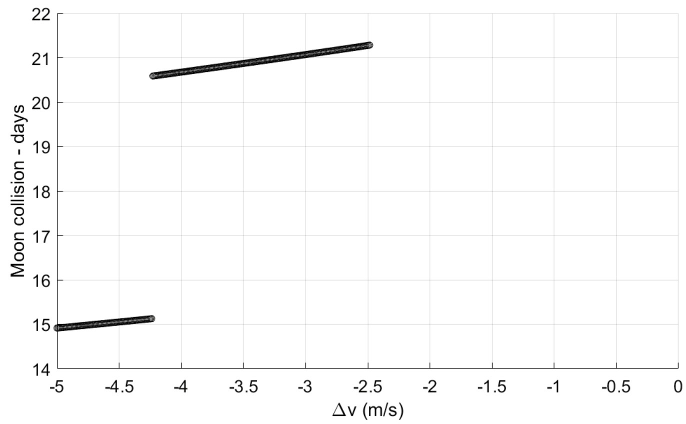

Figure 8: Time to collision with the Moon, after disposal from NRHO

CONCLUSION

The mission analysis for Heracles has been refined to take several new operational aspects into account, e.g. to limit the main engine firings due to the utilization of a cryogenic main engine. A launch date dependent mission timeline has been established, which also allowed to identify critical trades, e.g. between the size of the transfer correction manoeuvre after LLO departure and the time of manoeuvre execution, which could place heavy constraints on the available ground systems.

A further aspect that has been looked into is the disposal of the ascent stage after delivery of the samples. From the available options crashing the LAE on the moon seems to be the most attractive one, since the run-down phase is rather short and the disposal it permanent, while e.g. a heliocentric disposal would keep a small probability of the vehicle returning to the Earth-Moon system after several years.

Future mission analysis work will focus on the interface between GNC for the powered descent phase as well as for the rendezvous and docking phase. The available sensors will determine the requirements to delivered the S/C stack more or less accurately to a specific position with respect to either the landing site or the LOP-G for the start of the closed loop controlled mission phases.

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