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Suitability of reusability and in-situ propellant production for a Lunar transportation system

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Abstract

Sustainable continuous presence on the Moon is a long term goal of humankind. However, the few vehicles that were built until now for lunar missions were far from having the capability to achieve this goal. Their designs were mainly influenced by short term considerations and the available technological state of the art. Consequently relying on a classic expendable transportation system based on storable propellant will not allow going much further than where we arrived until today. A new transportation architecture is needed. In the frame of the ROBEX (Robotic Exploration under Extreme Conditions) project, new types of lunar architecture are under study. Modularity, re-configurability and flexibility are playing a major role. The transportation system is of course a very important aspect of future Moon missions, as it can represent a very large share of the mass to be launched from Earth. Reducing the share of the lunar transportation system within the total mass launched from Earth is the best way to reduce the cost of Moon missions and therefore increase the overall programme sustainability. The preliminary design of a Moon transportation system based on reusable vehicles running on LOx/LH2 and benefiting from lunar in-situ propellant production is being performed. This transportation system is made out of two types of vehicle. The Reusable Lunar Resupply Vehicle (RLRV) is a lunar single stage to orbit vehicle able also to land on the Moon. The Reusable Trans-Lunar Vehicle (RTLV) is a space-tug able to transport payloads from LEO to a cis-lunar orbit and back. Several cis-lunar orbits are considered and their influence on the RLRV and RTLV designs is discussed. Possible vehicle cooperation, such as in-space cryogenic propellant transfer is also studied. For both vehicles, a structural preliminary sizing is performed. For the RLRV in particular, the landing dynamic has been analysed to refine the landing loads. Based on these results, the potential gain brought by reusability to the Lunar transportation system is demonstrated for different cis-lunar staging orbits and operational modes.

Keywords: Moon, Lunar mission, transportation system, ISPP, reusability, LOx/LH2, Moon vicinity, NRO, DRO, EML1, LLO, landing gear, in-space cryogenic propellant transfer

Nomenclature

A_h	Horizontal Acceleration	[m/s²]
A_v	Vertical Acceleration	[m/s²]
CoG	Center of Gravity	[m]
D_c	Clearance Distance	[m]
D_s	Stability Distance	[m]
F_p	Footprint Radius	[m]
g	Lunar Gravity	[m/s²]
g0	Standard Earth gravitational acceleration	[m/s²]
Isp	Vacuum Specific Impulse	[s]
L_p	Primary Load Level	[-]
L_s	Secondary Load Level	[-]
L_v	Height of Landing Leg	[m]
MR	Engine Mixture Ratio	[-]
Pcc	Combustion Chamber Pressure	[bar]
T	Thrust	[kN]
V_h	Horizontal Velocity	[m/s]
V_v	Vertical Velocity	[m/s]
W	(Earth) Weight	[kN]
ΔV	Velocity Increment	[km/s]
θ	Landing Slope	[°]
μ	Friction Coefficient	[-]
τ_p	Primary Angle	[°]
τ_s	Secondary Angle	[°]

Acronyms/Abbreviations

API	Advanced Porous Injector
APU	Auxiliary Power Unit
CR3BP	Circular-Restricted 3 Body Problem
DRO	Distant Retrograde Orbit
DROI	DRO Insertion
E-M	Earth Moon
EML1	Earth Moon Lagrange Point 1
GH2	Gaseous Hydrogen
GOx	Gaseous Oxygen
IC	Internal combustion
IDSS	International Docking System Standard
ISG	Integrated Starter Generator
ISPP	In-Situ Propellant Production
IVF	Integrated Vehicle Fluids

L1OI	EML1 Orbit Insertion
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LM	Lunar Module of the Apollo programme
LOI	Lunar Orbit Insertion
LOx	Liquid Oxygen
LSAM	Constellation Lunar Surface Access Module
NPSP	Net Positive Suction Pressure
NRO	Near-Rectilinear Orbit
NROI	NRO Insertion
RCS	Reaction Control System
RLRV	Reusable Lunar Resupply Vehicle
RTLV	Reusable Trans-Lunar Vehicle
SDRO	Selenocentric Distant Retrograde Orbit
SSTO	Single Stage To Orbit
TOF	Time of Flight
TOI	Transfer Orbit Insertion
TRL	Technology Readiness Level
ULA	United Launch Alliance

1. Introduction

Only few years after the successful flight of Sputnik, and within only one decade a total of 18 soft Moon landings were performed successfully by spacecraft from the Soviet Union and the USA. This impressive result was driven by the Space Race between the USA and the Soviet Union. The main design driver was to be the first. It resulted in similar designs for all vehicles. They were expendable and based on storable hypergolic propellants carried from Earth. While storable propellants are particularly adapted to several day long missions and the hypergolic property increases the reliability of the mission, it implies the fuelling of the vehicle with relatively large mass of propellant coming from the Earth. The spacecraft were also optimized for very specific missions, with limited considerations about a possible future permanent presence on the lunar surface. In the context of the beginning of space exploration and the Space Race, such designs were the best answers to achieve the goals of that time.

Nowadays the design drivers are different. Cost and added-value for the society play a central role. For this reason, long-term planning and sustainability considerations are very important aspects. In 60 years of astronautics, numerous technologies had time to mature to allow novel concepts. This is the case of cryogenic propulsion. Others, such as reusability of Earth-based systems, are currently arriving on the market. All these evolutions are now making concepts feasible for which the development of In-Situ Propellant Production (ISPP) could also bring large synergetic benefits. Therefore, "brute-force means" [1] such as Apollo-type designs relying on very heavy lift launchers launching for every single mission all the required propellant and brand new vehicles should not necessary continue serving as reference for future Moon transportation systems.

Relying on heritage to reduce risks is important. A way to increase the sustainability of a Moon program would be to take advantage of the many new technological developments which have been achieved or made possible in the past years, rather than reusing old systems or subsystems. The initial investment might be higher if new subsystems are developed, but on a long perspective this can be much more economical. For instance, the understanding of cryogenic propellant management has increased a lot in the last fifty years, especially in the USA with the Centaur upper stage [2], but also in Europe with Ariane. This understanding is key to allow long duration missions with cryogenic propellants without too large losses. In the past, many concepts of long duration (several days) missions using cryogenic propellants considered as baseline the use of cryo-cooler. These active cooling devices would have pumped out from the propellant the heat coming from the solar radiation, to avoid propellant venting. Cryo-coolers are however complex, require external energy source and radiators [3]. In addition it still has a low TRL for such a kind of applications. Recently, multi-purpose concepts using the vaporized propellant in the ullage of the tanks have been proposed. It opens completely new possibilities for long term missions without cryo-coolers. For instance the Integrated Vehicle Fluid system proposed by ULA for its future ACES upper stage (see [4]) utilised a piston engine fed with the ullage gases which normally would be vented to avoid overpressure in the tanks. The piston engine is able to produce electricity and replace parts of the batteries, helps main engine re-ignition and propellant pressurisation, and allows large extension of the mission duration. Applying such a system for a Moon transport system would strongly ease the use of the high performance LOx/LH2 cryogenic propellant combination.

ISPP has not been tested yet on the Moon, but numerous studies on the topic have been performed and several solutions to produce LOx from Moon regolith exist and some of them have even been tested in laboratories or even during lunar analogue field tests [5]. According to Sanders [5], some of these ISPP methods such as hydrogen reduction of ilmenite would automatically allow to gather simultaneously water and interesting volatiles such as H_2 present in the regolith. Miliken et al. [6] recently estimated that pyroclastic deposits on the Moon contain larger water concentrations than previously estimated. This is particularly the case for areas with high titanium contents, which are precisely those where ilmenite (titanium-iron oxide) from which O_2 can be extracted, can be found. Some studies such as the one of Davidson et al. [7] considered ISPP but usually only for O_2 and not for H_2, as at this time the presence of water on the Moon was not known. Recent studies [8], in particular those analysing the data of the LCROSS mission [9], also indicate that water ice is present on the Moon in particular in polar regions. Although it is not clear yet what would be the best way to gather this water and how much water could be gathered, this would allow producing not only LOx but also some LH2. Source of propellant directly on the Moon would allow reducing strongly the mass to be launched from the Earth. This propellant could be useful in the vicinity of the Moon but also maybe closer to the Earth such as in LEO. The ΔV from the Moon's surface to LEO is indeed smaller than from the Earth's surface to LEO as shown in [10]. In other words, not only a combined Moon lander and Moon ascent vehicle could benefit from propellant production on the Moon, but also a transfer vehicle flying between LEO and a cis-lunar orbit. The combined lander and ascent vehicle could have an operational scenario similar to a SpaceX Falcon 9 first stage with the big advantage that the high loads due to an atmospheric re-entry would not exist. Reusing this part of a Moon transportation system is actually expected to be extremely interesting as it will avoid several heavy lift-launches from Earth. This vehicle, the Reusable Lunar Resupply Vehicle (RLRV) would also bring flexibility to the design. Indeed missions not requiring the full capability of the vehicle could be flown under conditions limiting the wear of the vehicle. It could even be used to transport payload or propellant from one point on the Moon to another through suborbital flights.

Reusability of a vehicle flying between LEO and a cis-lunar orbit, the Reusable Trans-Lunar Vehicle (RTLV), could further decrease the mass of payload to be launched from Earth. Actually, reusability is expected to be beneficial for the different sections of the transportation system: the Earth launch vehicle, RLRV and the RTLV. A possible operational scenario is shown in Fig. 1.

The choice of the cis-lunar orbit where the RTLV and the RLRV should perform rendezvous is also influencing strongly the whole system design. Rendezvous in Low Lunar Orbit (LLO), as for Apollo, is not necessarily optimal. Recently analyses have been performed to take advantage of orbits for which the influence of the Earth gravity is not negligible such as orbit about the Lagrange Point 1 (EML1), Near Rectilinear Orbit (NRO) or Distant Retrograde Orbit (DRO). These orbits have advantages in term of stability and/or Moon coverage. The repartition of the ΔV between the RTLV and the RLRV is also different, leading to different Moon transportation systems.

After comparing possible cis-lunar orbits for rendezvous and staging, the preliminary design of the RLRV is presented. In particular the landing dynamic of this vehicle is modelled in order to derive the corresponding landing loads. These results are then used to pre-size the vehicle and determine the correlation between RLRV propellant tankage and structural index.

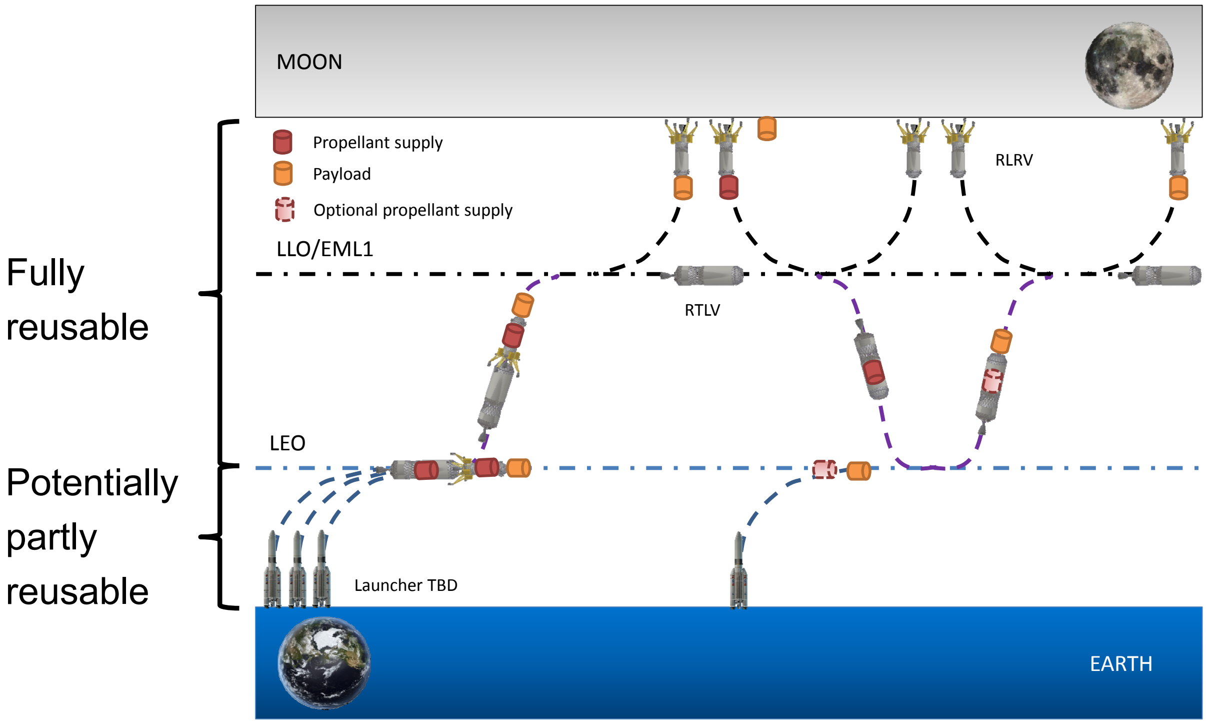

Fig. 1. Possible operational scenario for the proposed Moon transportation system

Similarly for the RTLV preliminary sizings are performed to establish a correlation between propellant tankage and structural index. For that purpose in addition of simple structural sizing, subsystems are pre-sized. In particular, the propellant transfer system concept is presented. These results are then used to assess the advantages brought by reusability.

2. Moon transportation system overview

2.1 Vehicle design rationale

Today's design proposals are still largely influenced by the experience gathered during the Space Race. This is logical, as it allows reducing the initial investment by reusing or re-adapting existing hardware or method. But this can only result in a classic design, not able to fully benefit from the advantages of new technologies that might be implemented.

In the frame of this study six ideas, which are expected to bring cost reduction and improvement in the sustainability of Moon missions with permanent presence of robots or humans, are combined:

- the use of turbo-pump fed cryogenic engines
- the use of ISPP
- the possibility to replace LLO rendezvous by other cis-lunar orbits
- the implementation of reusability
- the use of an advanced propellant management system
- the use of in-space propellant transfer

2.2 General approach for the preliminary design

After defining possible missions for the transportation system [11], and the general transportation system concept idea, a first version of the RLRV was pre-sized [10]. This preliminary version with a maximum propellant loading of 20 tons (H20), already considered loads encountered during the in-space flight, during the descent or during the launch from Earth. Some aspects, as for instance the landing dynamic was not considered in this first iteration loop. Similarly the influence of the RTLV on the RLRV, such as the required propellant for the RTLV refuelling was considered only very superficially and not for what concerns the payload and tankage capability from the Moon surface.

The second iteration loop, presented in the following chapters includes the influence of the landing dynamic on the choice of the architecture and the mass of the landing legs. Indeed the characteristics of the landing gear: size and mass depends strongly on the position of the centre of gravity and velocity during landing. While longer and slender tanks lead to a lighter structure, they push the centre of gravity higher and thus decrease the stability at landing. Larger landing legs are then required to stabilise the vehicle. A trade-off between tank architecture and required landing gear size has to be performed to keep the whole mass low. Analyses pursuing this goal are presented in section 5.1. The second design iteration also refines the propellant transfer procedure and systems. In particular, an interface concept named ADAPT for Advanced Docking And Propellant Transfer is proposed and described in section 6.2.4. Engine alternatives have also been considered, see section 4. The structural model of the RLRV has been refined with the consideration of landing loads and a structural model of the RTLV has been established. The structural design was optimised for both vehicles across a large propellant loading range in order to derive structural index trend lines. These trend lines give for the preliminary design sufficient information to get a good idea of the whole transportation system size.

3. Staging in the vicinity of the Moon

The staging between the RLRV and the RTLV should be performed in the Moon vicinity. The Apollo transportation system was based on a staging between the descent stage and the transfer stage in LLO. The rendezvous between the Apollo Ascent module and the Command module which brought the astronauts back to Earth also occurred in LLO.

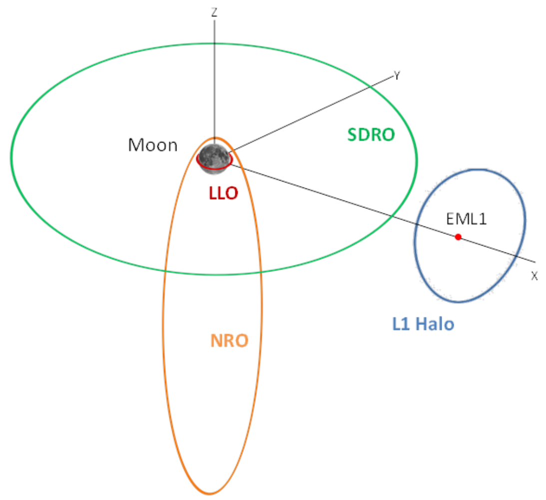

Fig. 2. Overview of considered cis-lunar staging orbits

Many transportation system concepts designed after the Apollo program to return to the Moon, proposed therefore LLO staging as a baseline. However in the past years the interest for alternative staging locations/orbits in the Moon vicinity has increased. Each of these locations/orbits has its own advantages and drawbacks.

An overview of the characteristics of potential cis-lunar staging orbits (see Fig. 2) is proposed in the following subsections. Compared characteristics are the orbital period of the orbit, the ΔV to reach the orbit from a 200 km LEO and the associated the time of flight, the

ΔV to reach the orbit from a LLO and the associated time of flight. Note that we are considering freight transportation mission and the influence on the ΔV of human spaceflight specific manoeuvres is therefore disregarded.

3.1 Candidate staging orbits

3.1.1 Low Lunar Orbit (LLO)

For the Apollo program, such orbits were used as phasing orbits for rendezvous and for surface access. The considered LLO is a 100 km circular lunar orbit, with a low inclination. Transfers from LEO to LLO have been computed with GMAT Version R2016a (see Fig. 3) in order to determine the required ΔV and TOF. Note that these values depend on the date of departure. Based on GMAT results, it appears that for most departure dates, the ΔV variation is about 50 m/s and the TOF varying between 3.5 and 5 days.

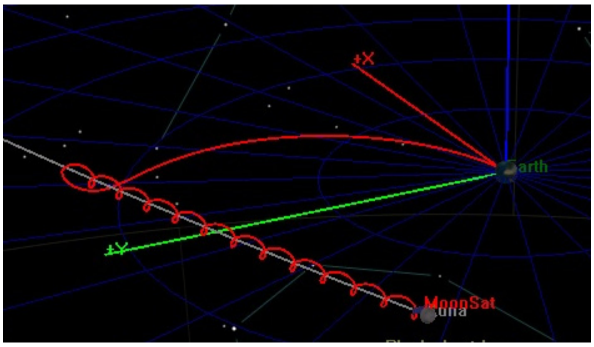

Fig. 3. GMAT Simulation of a LEO-LLO transfer for a departure on January 1st, 2015, 11:59 UTC (Earth-centred inertial frame of reference)

The values obtained in GMAT (see Table 1) are consistent with those that can be found in the literature [12] and [13]. Note that the ΔV needed for the descent is not considered here. For example the Moon landing performed during the Apollo mission has recorded a ΔV of 2130 m/s for the descent from the LLO to the lunar surface [14]. The ΔV can actually vary in function of the thrust to mass ratio of the vehicle and the mission itself. Manned mission may require more ΔV for safety reasons. Results of [10] show that if the thrust to mass ratio varies in a realistic range, the ΔV from LLO to lunar ground could vary from 1780 to 2200 m/s. This variation is explained by the influence of the gravitational losses which are functions of the flight path angle during the descent/ascent and therefore of the thrust to mass ratio of the vehicle. In order to compare the cis-lunar staging orbits independently of the vehicle design the LLO is chosen as the point of comparison for every other orbit. Flying to the Moon through LLO has almost no incidence on the ΔV and TOF, even if the staging is not occurring in LLO.

3.1.2 Halo orbit around the Earth-Moon Lagrange point 1 (EML1)

Halo orbits around the Earth-Moon Lagrange points are being extensively studied [12], [13], [15] and [16] for their mild instability leading to small ΔV for station-keeping. For each state along the orbit, there exists, due to the gravitational interaction of the Earth and the Moon, an unstable and a stable manifold along which a vehicle would drift if no station-keeping is performed. Using manifolds allows travelling at reduced ΔV but slowly, according to [12] and [16]. The manifolds are particularly adapted when the ΔV minimisation is the main trajectory selection criterion. In the case of vehicle using cryogenics propellants such as LOx and LH2, the flight duration should be kept relatively short to avoid too much propellant losses. Halo orbits seem also very interesting for the establishment of permanent spacecraft in the vicinity of the Moon such as propellant depots and manned space stations [16] and [17]. The characteristics of transfers to and from this EML1 halo from/to the Earth and the Moon have been computed both with GMAT and a Matlab code developed at ISAE Supaéro [18]. This code considers the Earth-Moon system as a CR3BP and its working principle is displayed in Fig. 4.

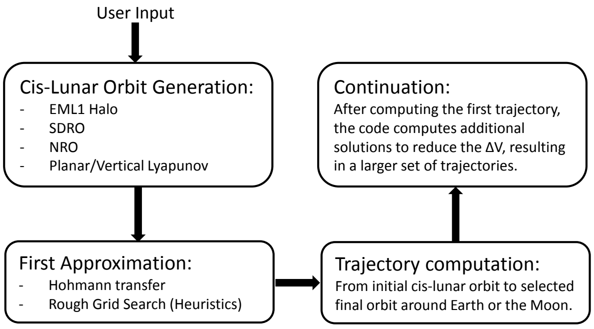

Fig. 4. Flowchart of Supaéro's Matlab code working principle

This code has also the advantage to allow quick parametric analyses. The ephemerides model used by GMAT allows computing more accurate trajectory but is less adapted for parametric studies. Results for transfer from LEO to halo orbit around EML1 are summarized in Table 1. The trajectory computed with GMAT is displayed in Fig. 5.

Results obtained with the Supaéro code show a very good agreement with those obtained with GMAT. The exact values of ΔV and TOF calculated with GMAT depend actually on the ephemerides and on the chosen halo orbit. These ranges are also in accordance with [12] and [13].

For what concerns the transfer from halo orbit around EML1 to a 100 km LLO computations have been performed only with the Supaéro code (see Fig. 6) and the results are summarized in Table 1. One can observe large variations of ΔV and TOF that correspond to the fact that the transfer between the halo orbit and the LLO can be performed differently depending if it is required to save fuel by reducing the ΔV or to have quick transfer, reducing the TOF by increasing the ΔV. A study that has been conducted on the last version of the Matlab code seems to show that there is a direct relation between the ΔV and the TOF. Reducing one implies increasing the other.

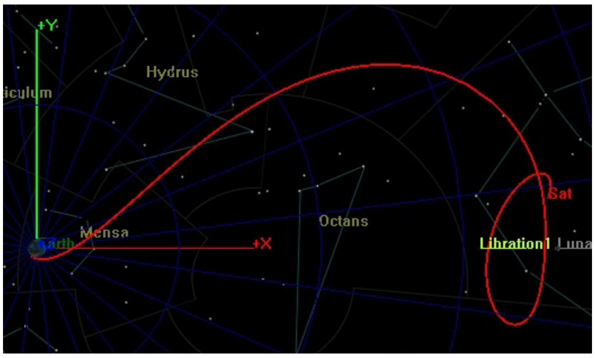

Fig. 5. Transfer from LEO to EML1 halo computed with GMAT for a departure on January 1st, 2015, 11:59 UTC (Earth-centred inertial frame of reference)

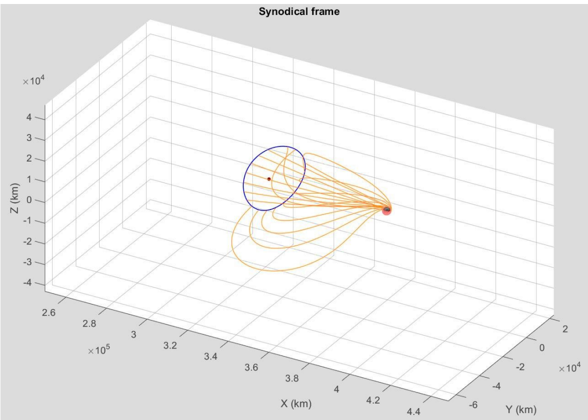

Fig. 6. Resulting possible Hohmann-like transfer between a LLO and an EML1 halo orbit, as computed with Supaéro's Matlab code.

The results of the Supaéro code for the LLO-EML1 transfer are consistent with the other references such as [13]. Two types of trajectories have been considered for vehicle design presented in section 7. The first one has a low ΔV and a relatively high TOF (ΔV = 0.8 km/s and TOF = 3 days), which we call "slow EML1-LLO transfer". The second one, the "fast EML1-LLO transfer", requires only a time of flight of 0.8 days but a ΔV of 1.5 km/s. In case of cryogenic propellant system the faster trajectories might be advantageous, if the propellant boil-off exceeds the additional propellant required for a higher ΔV.

The Matlab code also allows computing the trajectories, ΔV and TOF for planar and vertical Lyapunov orbits about EML1. The corresponding ΔV and TOF are very close to those found for halo orbits. Therefore Lyapunov orbits are not shown separately but remain as possible candidates for the cis-lunar staging.

3.1.3 Selenocentric Distant Retrograde Orbits SDRO

Selenocentric distant retrograde orbits are a specific kind of orbit around the Earth which appear to orbit circularly and in retrograde direction around the Moon. These orbits can be computed in the CR3BP model and appear as orbits around the Moon in the Earth-Moon orbital plane. SDRO are considered to be very stable orbits (up to 100 years, mainly due to their constantly high altitude) which limits strongly the station keeping requirements. These orbits are being considered for instance for the asteroid redirection mission of NASA [19] and also for connected future manned flights within the Orion program. Supaéro's Matlab code allows studying transfers from a SDRO to the LLO and from the LEO to the SDRO. For the transfer from LEO to SDRO, a far side insertion has been selected as it is the manoeuvre that features the lowest ΔV. The results of the study prove to be similar to the one found in the literature [20] and [21] though with a slightly higher ΔV. The characteristics of a flight from LEO to SDRO can be summarized as in Table 1.

A simplified estimation of the required ΔV for a transfer from the Moon surface has been performed with the help of a Hohmann transfer by Adamo et al. [20]. It has been shown that the required ΔV is between 2.5 and 2.6 km/s and the TOF will depend heavily on the size of the SDRO. The TOF will vary from 84 hours for a 70000 km SDRO to 5.6 hours for a 10000 km one [20]. Even smaller SDRO are possible and they will require shorter TOFs but the ΔV will still be in the range of 2.5 to 2.6 km/s. For a transfer from a SDRO to a polar LLO, Whitley et al. [13] have estimated a ΔV of 830 m/s. The Supaéro's code seems to show similar results in terms of ΔV and TOF but offers a broader range of possible solutions. The computing was done for a 70000 km SDRO towards an equatorial LLO at 100 km. The range of realistic solutions goes from a ΔV of 1500 m/s for a TOF of 1 day to a ΔV of 800 m/s for 3.5 days of travel. The computation has also been performed for a polar LLO and it corresponds roughly to the same order of magnitude. These values are summarized in the Table 1. The study has also been done for smaller SDRO in order to assess the impact of the SDRO size. With a 10000 km SDRO, the TOF is drastically reduced to 5.6 hours, the ΔV does not change. However one of the drawbacks of smaller SDROs is that the access to high inclination becomes very costly. Small SDROs are very similar to high Lunar orbits, the access cost and the TOF are low but changes of inclination are prohibitive.

Table 1. Overview of potential cis-lunar staging orbit characteristics based on Matlab code, [10], [12], [13], [14], [20], [21] and [23]

	LLO	Halo EML1	SDRO	NRO
Orbit period	~2 hrs	11-12 days	0.2-12 days	6-8 days
Amplitude range	100 km	1000-20000 km	10000-70000 km	2000-75000 km
Orientation wrt E-M plane	Any inclination	Size-dependent	Equatorial	Roughly polar
Stability	< 2 months without station keeping	5-50 m/s/yr for station keeping	>100 years without station keeping	5 m/s/yr for station keeping
Coms with Earth	Some LOS	Permanent	Short LOS	Permanent
Access to the orbit from a 200 km LEO (ΔV in km/s)				
ΔV Transfer	3.14	3.11	3.15	3.13
ΔV Insertion	0.81	0.62-0.83	0.58-0.68	0.42
ΔV Total	3.95	3.73-3.94	3.75-3.83	3.55
TOF (Days)	4	4-4.5	3-6.2	5
Moon surface access from the diverse orbits (ΔV in km/s)				
ΔV to equatorial LLO*	0	0.75	0.8-1.5	0.9
ΔV to Polar LLO*	0**	0.8	0.88-1.5 (only for 70000 km)	0.73
ΔV to Lunar Surface***	~1.9	2.65-2.7	2.6-3.4	2.63-2.8
TOF (Days)	0	3	0.2-3.5	0.5

* LLO (circular 100 km) is the standard phasing orbit for surface access. A LLO of specific inclination is a good intermediate step on the way between the staging position and the lunar surface.
** The inclination change on LLO would be very costly and is not relevant as the final LLO will be chosen for a specific landing site
***Transfer through a LLO considering a ΔV = 1.9 km/s for the descent from LLO to lunar surface [10].

More detailed studies are currently conducted on the SDRO notably by the research team of ISAE Supaéro.

3.1.4 Near-Rectilinear Orbit (NRO)

Near Rectilinear Halo Orbits constitute a type of orbit that results from solving numerically the CR3BP. While appearing much like a highly elliptic polar orbit in the Moon-corotating frame, these orbits are some kind of intermediary orbits between EML1 and EML2 halo orbits in the Earth-Moon system. They can be considered repetitive gravity-assists resulting in a 1:1 synchronous orbit with the Moon around the Earth. As described in [13] and [22], they are characterized by a very high amplitude over one lunar pole, where similar to an independent inclined orbit around the Earth and a very low amplitude over the opposite pole, the fly-by closest approach. These orbits appear as highly elliptical polar orbits around the Moon in the CR3BP model. NRO are being studied in the frame of the future Orion mission by Whitley et al. [13]. An overview of LEO-NRO transfer based on data from [13] and [22] can be found in Table 1. The transfer from a LEO to a NRO is very similar in ΔV and TOF to a classical transfer from LEO to LLO, however the spacecraft will perform a fly-by around the Moon before inserting into the NRO which reduces slightly the ΔV of the insertion.

The transfer from a NRO to a LLO cost an additional ΔV of 0.7 to 0.9 km/s [13], as seen in Table 1, depending of the final orbit inclination. NRO is particularly well adapted for missions towards polar regions. Actually the NROs belong to the halo family, but as they approach the Moon closely, they are less instable than classic halo orbits. The time of flight is shorter than for halo but the ΔV is in the same range, as long as manifolds are not directly used.

3.2 Rendezvous capabilities:

An important aspect for the choice of an interesting cis-lunar staging orbit is the ability to perform rendezvous between the RTLV and the RLRV.

3.2.1 LLO:

The rendezvous and docking in LLO is not a new problem and has been successfully performed several times during the Apollo program of NASA [14]. The relative instability [23] of the LLO is not a real problem if the spacecraft does not stay long on this type of orbit (less than a month). If a long duration is required, a frozen Moon orbit can be selected [23].

The rendezvous and docking manoeuvres will be quite simple and can be automated the same way as LEO operations with the ISS for example. A double co-elliptic trajectory could be appropriate for this kind of mission.

3.2.2 EML1 halo orbits:

The rendezvous and docking operations for a halo orbit around a Lagrange point is not a fully solved process nowadays. The approach and docking manoeuvres (less than 50 km from the target) seems not to pose any problem and could be handled in a similar way as for LEO [23]. However the rendezvous and phasing phase is problematic due to the slow dynamic of the halo orbits (Halo orbits have an about 12 days period) around the Earth-Moon libration points. This period is not much influenced by the orbital amplitude.

According to [16] the rendezvous of spacecraft on halo orbits is possible by basically transferring the chaser from a nearby halo orbit to the target's halo orbit using the invariant manifolds of these orbits. In the simulations performed by Lizy-Destrez et al. [16], the operation would take 3.63 days and would bring the chaser to 42 km distance from the chaser which is ideal for starting the approach and docking operations. The total ΔV required would also be very low: less than 2 m/s. The problem with this method is the relatively long transfer time, which might not be optimal for a cryogenic vehicle. The almost four days added with the time necessary to fly from the Moon to the halo orbit, sum up to more than 6 days. Quicker rendezvous at the cost of a higher ΔV as proposed in [16] would be preferable.

3.2.3 SDRO:

Few studies have been conducted on rendezvous and docking on a DRO. In the frame of NASA's asteroid redirect mission Hinkel et al. [16] show that the rendezvous and docking on a SDRO does not seem problematic and could be done with a small amount of fuel and in a short time.

Current estimations are considering the Orion spacecraft to rendezvous with the captured asteroid on a 70000 km SDRO. In order to reduce the distance between the two spacecraft, the chaser will perform a simple two-burn manoeuvre. Required ΔV and TOF required for the rendezvous depend from each other [24]. If the TOF is critical, it can be reduced with more ΔV intensive manoeuvres. Currently, the anticipated navigation errors should allow the chaser to be in the 30-50 km range from the target after insertion in the SDRO [16]. Given 6 hours to perform the whole rendezvous, the total ΔV should be about 5 m/s.

When the spacecraft are less than 2 km from each other, the LIDAR of the chaser should acquire the target and docking operations as in LEO can be performed. The duration from the approach from the 660 m position to the docking confirmation should be less than two hours which would give a TOF of 8 h for the whole rendezvous and docking operations.

3.2.4 NRO:

As for SDRO, very few studies have been conducted concerning the rendezvous and docking in NRO. This orbit is being considered as a possible candidate for the Orion mission around the Moon [13]. As NRO are in the halo family it is expected that rendezvous can be performed in a similar way as for classic halo orbit. Work is currently being performed on this topic [25].

4. Engine selection

Preliminary engine system designs have been performed with the help of the DLR in-house tool 'lrp'. The 100 kN expander cycle engine presented in [10] was taken as reference and the expansion ratio has been varied. In particular using large expansion ratio for the RTLV is particularly promising. On the RLRV, the nozzle should not get too long in order to guarantee sufficient ground clearance with a reasonably long landing gear. Another impact from the landing is the requirement to have a thrust to weight ratio around one just before the touch-down. On the other hand as shown in [10], it is important to have a relatively high thrust during the descent or the ascent to reduce the gravity losses and consequently the ΔV. A thrust level leading to an Earth acceleration of 0.5 g_0 at beginning of the ascent or descent seems optimal for the reference mission with a 10 ton payload. Further increasing the thrust is almost not decreasing the ΔV, while the engine mass increases. Decreasing the thrust leads to an increase of the ΔV which the decrease in engine mass cannot compensate. To deal with these two contradicting requirements of a low final landing phase thrust and high ascent and descent maximal thrust, throttleability is required. If only one engine is used a relatively deep throttling (less than 10%) would be needed in most cases. In case, such a deep throttling cannot be easily achieved or is implying too large losses, several smaller engines can be used at reasonable mass increase. In order to keep a good level of performance within the throttling range, specific measure can be taken at engine design. A special care should be attached at the injection method. Advanced porous injector (API) [26] could be for instance a good solution, as it is characterised by a good efficiency over a large range of throttle level. In such a way the full thrust performance are not too much impacted by the throttling capability.

The RLRV and the RTLV should be reusable vehicles. Concerning the reusability of rocket propelled vehicles, the engine is one of the most critical components. Technical choices at engine design level have a large influence on the life time and the number of cycle that an engine can withstand. Amongst others, important parameters are the combustion chamber pressure and the turbine inlet temperature. Recent analyses [27], in particular, showed the influence of the engine cycle on the reusability, through the implication that the chosen engine cycle has on the cooling channel pressure and temperature and on the turbine inlet temperature. As a consequence, gas generator engines which appear to have some advantages over expander cycle engine can be still considered as a possible alternative. Moreover, in case of a required single engine thrust level reaching levels over 150-200 kN, the mass penalty for expander cycle with respect to gas generator is not necessarily compensated by the higher specific impulse.

Specific impulse results from the simulation are presented in Fig. 7. In particular, the advantage of 3 to 4 s in specific impulse for the expander cycle over the gas generator cycle at a 60 bar combustion chamber pressure and various expansion ratios can be observed. Increasing the combustion chamber pressure for the gas generator cycle leads to lower specific impulse, as proportionally more propellant is needed by the gas generator, but in the same time the design is getting more compact and lighter with increasing chamber pressure. As a consequence for a given mass or a given engine length, the specific impulse of the 80 bar combustion chamber version is higher than for the 60 bar combustion chamber version.

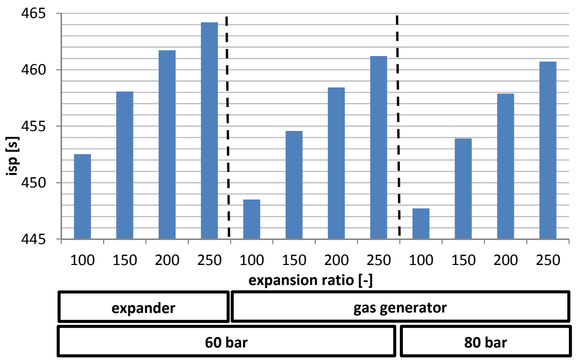

Fig. 7. Computed specific impulse for potential expander and gas generator cycle LOx/LH2 engines

Considering the expected intrinsic higher life duration of gas generator cycle engines with respect to expander cycle engines [27], the 80 bar version of the gas generator engine can be compared directly to the 60 bar expander cycle engine. It then appears that for instance a 100 kN expander cycle engine with an expansion ratio of 100 is approximately as long and as heavy as a 80 bar gas generator engine with an expansion ratio of 200. Consequently gas generator engines with 80 bar combustion chamber pressure are taken as baseline for the sizing of the RTLV and of the RLRV in the following sections. Further analyses should help confirming this choice. But a relatively long lifetime and a high number of cycles can probably be achieved. For instance: Vulcain 2 engine tests demonstrated a lifetime of over 10000 seconds during 20 cycles, in spite of a higher combustion chamber pressure of about 120 bar [28].

In order to reduce costs, synergies have to be used between the RTLV and the RLRV. For this reason, the baseline design considers the same engine for both vehicles. The only difference concerns the nozzle expansion ratio which is taken larger for the RTLV. In the current design, it is considered that both vehicles are equipped with only one engine. Some preliminary results however show that it might be advantageous under particular conditions to use several engines for the RLRV and only one for the RTLV which is less impacted by the thrust level. This option will be considered in future work.

5. RLRV

The preliminary design of the RLRV presented in [10] has been refined. An important improvement in the modelling is related to the landing gear system, which has been the object of a preliminary sizing based on landing dynamic simulations, see section 5.1. These results allowed to take into account with a better accuracy load cases associated to the landing and to generate for a large range of propellant tankage, a preliminary structural design and derive a structural index curve needed for the transportation system sizing: see section 5.2.

5.1 Landing gear system

The main function of the landing gear system is to provide energy absorption during touchdown to guarantee stability and adequate ground clearance to the lunar landing vehicle. The landing system must also attenuate the landing loads and minimise load accelerations applied to the lander to prevent damage on the structures or on-board components.

5.1.1 RLRV configurations

Different tank configurations of the RLRV have been considered in previous analyses [10]. The mass and height of the RLRV are important parameters to consider in terms of landing stability. Compact and short configurations with large but short tanks are much preferred for stability as compared to slender and tall configurations. Among all the tank configurations presented in [10], H20d3d3 (20 tons propellant tankage; 3 m diameter LH2 tank ; 3 m diameter LOx tank) and H20d4d3 (20 tons propellant tankage; 4 m diameter LH2 tank ; 3 m diameter LOx tank) configurations were analysed for landing stability. As expected, results show that H20d4d3 exhibits better landing stability characteristics than H20d3d3.

Table 2. Lander configurations

Parameter	Unit	Config. 01	Config. 02
Architecture	[-]	H20d4d3	H20d4d3
Payload mass	[kg]	25000	10000
Total mass	[Mg]	28.0	13.0
CoG height	[m]	14.97	13.02

In the following only results for the H20d4d3 configuration of the RLRV are presented. Payload of up to 25 tons could be carried during descent in the possible missions of the RLRV. Two landing configurations are considered in the preliminary landing gear design as shown in Table 2.

5.1.2 Landing system configuration

Amongst the several landing systems that have been proposed in the past, two main configurations can come into consideration for the RLRV: cantilever design and inverted tripod design, as shown in Fig. 8.

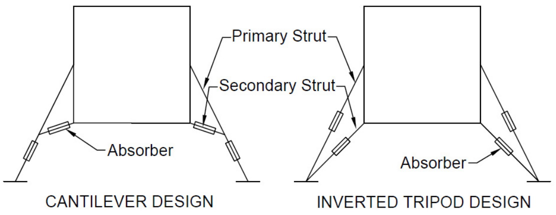

Fig. 8. Different landing gear configurations

The inverted tripod design has both the primary and secondary struts connected to the footpad while the cantilever design has the secondary strut connected to the lower end of the primary strut upper section.

Both designs have previously been compared in [29]. The cantilever design is chosen over the inverted tripod design, primarily because of the lighter structure since the secondary struts are much shorter. The connection of the secondary strut to the primary strut also reduces the risk of interference with obstacles in the vicinity of the footpad as compared to the inverted tripod design. A four legged landing system shall be adopted for the preliminary design.

5.1.3 Design drivers and parameters

The main design drivers which affect the preliminary design parameters of the landing system are the stability distance and ground clearance, as illustrated in Fig. 9.

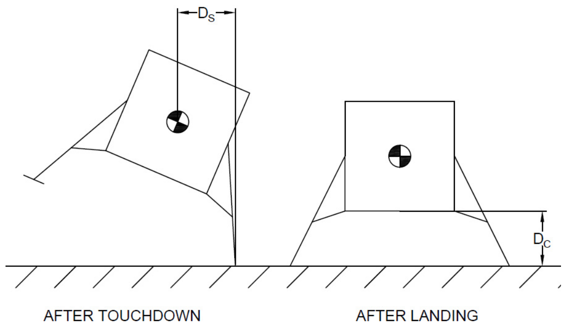

Fig. 9. Landing gear design drivers

Stability distance, D_s, is required to prevent toppling of the lunar vehicle and is measured by the minimum distance perpendicular to the gravity vector from the centre of gravity to the position of the footpad during the entire landing period.

Ground clearance, D_c, is measured from the interface between engine and the lander body to the ground. Sufficient ground clearance is necessary to allow for possible boulders or uneven surface during descent and engine performance to be unaffected at lift-off.

Design parameters, as shown in Fig. 10, are chosen to define the sizing of the landing gear system fulfilling the requirements in term of design drivers. These parameters will determine the structural and mass sizing of the landing gear initial design.

F_p is the horizontal distance from the center of gravity to the footpad. τ_p is the angle between the primary strut with the body vertical reference. τ_s is the angle between the secondary strut and the body vertical reference. L_p and L_s are the limit loads of the primary and secondary strut absorber elements respectively. The absorber elements absorb energy during landing by stroking at the limit loads, representing an ideal dynamic energy absorption.

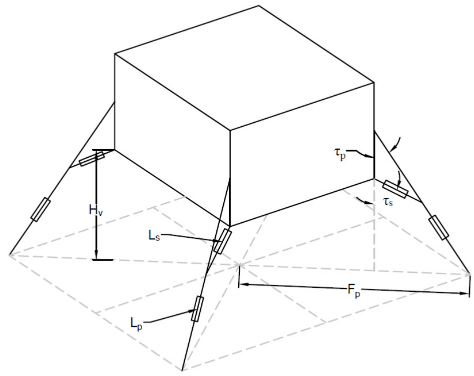

Fig. 10. Landing gear design parameters

5.1.4 Touchdown conditions

The touchdown conditions have a major impact on the landing dynamics which affect the design parameters of the landing system. In the case of the RLRV mission, after the first landing, landing operations should occur on a prepared surface. Hence, these landing conditions differ from the values which have been analysed for past landers such as Apollo lander or the Viking lander.

Gravity of the Moon is approximately 1/6 that of the Earth [30]. Prepared and from ground selected landing areas are assumed to be free of obstacles and not steeper than 2° (load case 1). Landing slope, θ, on first landing should not be steeper than 5° (load case 2). Friction coefficient, μ, is estimated to range from 0.3 to 0.9 as used in [31]. the vertical and horizontal velocities: V_v and V_h are estimated based on current landing technology precision.

Since the normal operations of the RLRV are expected to be carried out on prepared landing areas most of the time, the landing system should be able to land the RLRV with a payload up to 25 tons for load case 1. In the case of the initial landing in load case 2, the payload of the RLRV is limited at 10 tonnes in order not to overdesign for the load case 2 only and thus penalised the vehicle for subsequent landings. The other landing characteristics considered for both load cases are illustrated in Fig. 11.

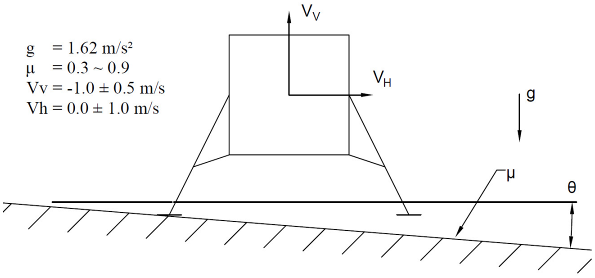

Fig. 11. Touchdown parameters

The landing orientation also influences the stability and ground clearance of the landing. Referring to Fig. 12, 2-2 landing orientation has a much smaller footprint radius compared to a 1-2-1 landing orientation, which will affect the stability. On the other hand, 1-2-1 landing orientation affects more of the ground clearance as landing on the first landing leg with maximum lander mass will result in larger stroking of the absorber before the other three legs touch the ground. In this study, 1-2-1 landing orientation will be analysed for load case 1 and 2-2 landing will be analysed for load case 2 for a conservative approach.

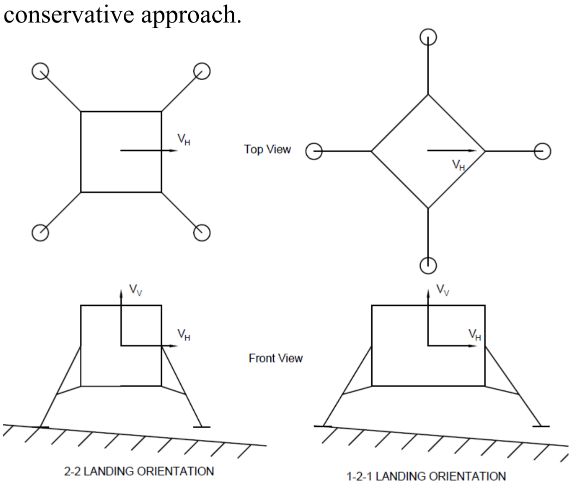

Fig. 12. Landing orientations

5.1.5 Model and simulation

A two dimensional cantilever-type lander model is modelled and simulated in the MATLAB/Simulink environment to assess the touchdown dynamics and determine the required design parameters.

Design parameters were optimised for the two load cases to analyse the touchdown performance based on the stability and ground clearance.

Based on [31], different terrain conditions have been studied for a preliminary approach. Landing away from a slope has a higher tendency of toppling over as compared with landing towards the slope. Higher friction coefficients will lead to a higher toppling risk while lower friction coefficients might contribute to a lower ground clearance.

Previous analysis also shows that certain design parameters can be predetermined at an estimated value. H_v is set at 4.0 m to allow clearance for the engine nozzle and exit exhaust gases. τ_s is set at 80° to provide an effective hold back on the spread of the primary struts. Both L_p and L_s are kept constant to simulate an ideal energy absorption. Hence, these design parameters can be kept at predetermined values while the footprint radius and primary angle is varied.

Fig. 13 and Fig. 14 show the effects of the footprint radius and primary angle on stability and ground clearance for load case 1. Negative stability distance indicates instability toppling over during landing and ground clearance below 3.1 m indicates either instability or insufficient clearance after landing.

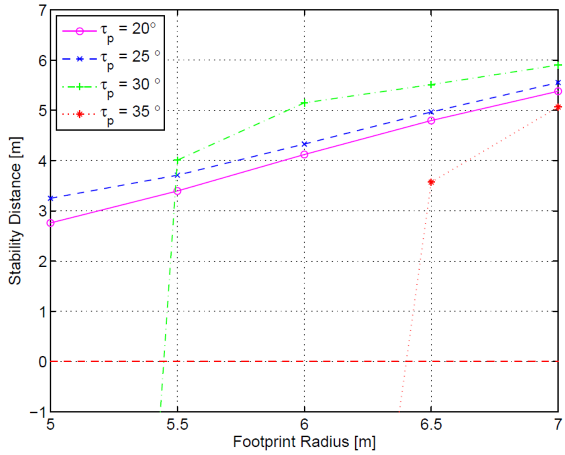

Fig. 13. Effect of footprint radius and primary angle on stability for load case 1

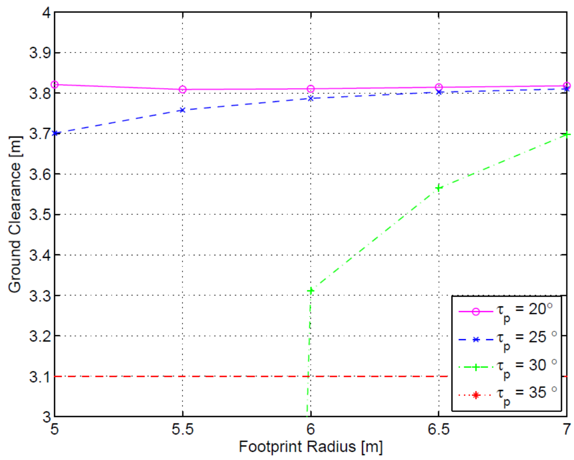

Fig. 14. Effect of footprint radius and primary angle on ground clearance for load case 1

As shown in Fig. 13, increasing footprint radius provides better landing stability as expected and increasing primary angle seems to have a better influence on the landing stability for primary angle of 20°, 25° and 30°. Fig. 14 shows that smaller primary angle is preferred for ground clearance.

Fig. 15 and Fig. 16 show the effects of the footprint radius and primary angle on stability and ground clearance for load case 2. Increasing footprint radius contributes to stability and lower primary angle provides better ground clearance, similar to results in load case 1.

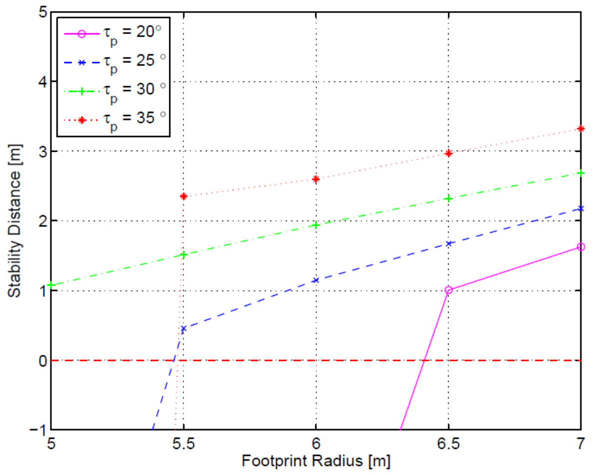

Fig. 15. Effect of footprint radius and primary angle on stability for load case 2

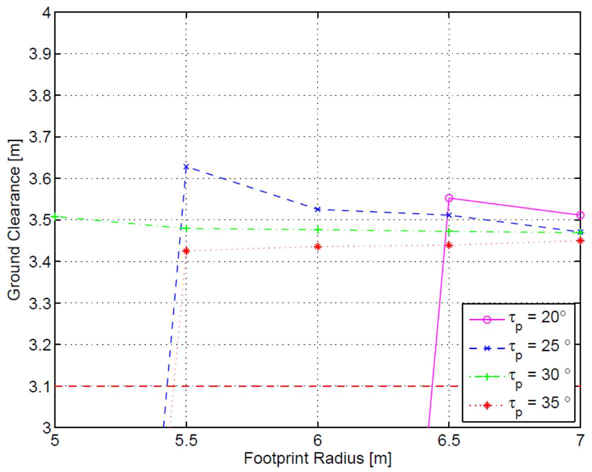

Fig. 16. Effect of footprint radius and primary angle on ground clearance for load case 2

It might be obvious that selecting a larger footprint radius and smaller primary angle will provide great stability and ground clearance. However, the mass of such configuration would increase tremendously which would not be desirable.

As a result, the preliminary design parameters values are selected based on a compromise between the landing performance and configuration mass. Robustness and adequate design margins are also taken into account in the selection to allow changes to be altered when further details are added to the landing gear design in the future. Table 3 shows a summary of the chosen parameter values.

Table 3. Selected preliminary design parameters

	Units	Value
Radius of footprint	[m]	5.5
Height of landing leg	[m]	4.0
Primary angle	[°]	25
Secondary angle	[°]	80

With the preliminary design parameters determined, the load-stroke curves of the primary and secondary absorber elements are re-designed to accommodate both landing load cases. Load levels are derived from previous study and stroke values are adjusted to meet the stability and clearance requirements. Adequate stroke is available for absorption without bottoming of the absorber. One possible design of a two stage load level curve, defined in each primary and secondary absorber element is shown in Fig. 17.

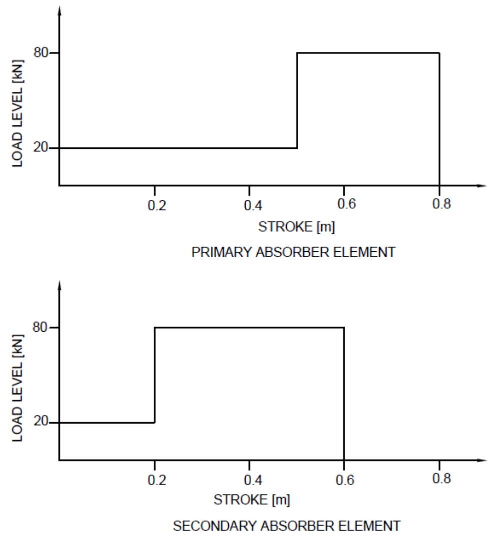

Fig. 17. Primary and secondary absorber element load-stroke curve

Table 4. Landing performance

Parameter	Unit	Load Case 1	Load Case 2
D_s	[m]	4.3	0.5
D_c	[m]	3.2	3.6

With this absorber element load-stroke curve design, the load cases are re-simulated and the landing performance is tabulated in Table 4.

5.1.6 Structural and mass sizing

With the selected design parameters, the mass of the landing legs can be estimated based on structural analysis of the loads acting on the struts of the landing legs and the length of the struts. Margins are considered on top of the structural sizing to account for the additional fittings, details and unaccounted masses at this stage of the mass estimation. The Apollo lunar lander landing system [29] has been studied closely to determine these margins. The preliminary design of the landing system is illustrated in Fig. 18.

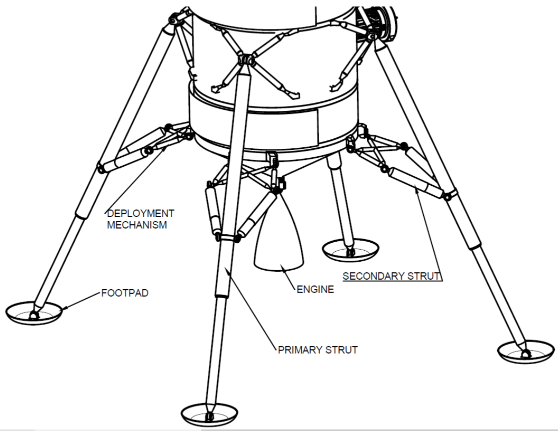

Fig. 18. RLRV landing gear system

The primary struts are mainly axial-loaded compressive strut subjected to bending moment due to the tension from the secondary strut during landing. The structural sizing of the primary strut is based on a structural safety factor of 1.5.

The secondary struts are axial-loaded struts with minimum lateral loads. The structural sizing of the secondary strut is based on a structural safety factor of 1.5.

The footpad is designed for a maximum penetration depth of 0.15 m of unprepared lunar regolith of bulk density 1.50 ± 0.05 g/cm^3 [30].

The deployment mechanism holds the landing legs stowed during launch and in-space propellant transfer operations. The mass is estimated proportionately to the secondary strut since the axial loadings on the deployment truss are similar to that of the secondary struts.

The thermal insulation mass is estimated based on the external surface area of the primary struts, secondary struts and footpad. Assuming a similar thermal insulation as on Apollo Lunar Lander [29], the mass per surface area is estimated to be about 0.6 kg/m^2.

Mass of the absorber elements within the core of the primary and secondary struts depends on type of shock absorber to be used. For this study, the mass of absorption elements is sized with aluminium honeycomb crushable elements. From the load-stroke curve in Fig. 17, first stage core density of the primary absorber element is estimated from [32] to be 32 kg/m^3 and that of the secondary absorber element is 64 kg/m^3. The second stage core density of the primary absorber element is estimated to be 64 kg/m^3 and that of the secondary absorber element is 112 kg/m^3.

Aluminium 7178 and typical CFRP are both considered as candidate materials for the primary, secondary struts and deployment truss. Aluminium 7178 was used in the primary struts of the Apollo Lunar Lander [29]. CFRP is widely used in both aviation and space industry and is a potential material to be considered for further mass reduction.

The resulting preliminary mass break-down for both materials is shown in Table 5. An important mass reduction can be achieved when using CFRP compared to aluminium.

Table 5. Preliminary mass summary of the landing system

	Unit	AL7178	CFRP
Primary Struts	[kg]	725.9	386.0
Secondary Struts	[kg]	36.5	20.9
Footpads	[kg]	38.6	38.6
Deployment Mechanism	[kg]	42.8	24.5
Thermal Insulation	[kg]	25.1	25.1
Absorber Element	[kg]	21.5	21.5
Total Landing System	[kg]	890.4	516.6

5.1.7 *Reusability of Landing System*

It is important to understand that although the energy absorber element in the previous section is sized by the honeycomb crushable type absorber, it is not entirely suitable for the reusability of RLRV. The usage of crushable material as energy absorber is possible only if the absorber element can be replaced for the next mission. Hence, it requires manpower or robotics involvement. Additionally, it can be complex and logistic intensive to replace the absorber cartridges.

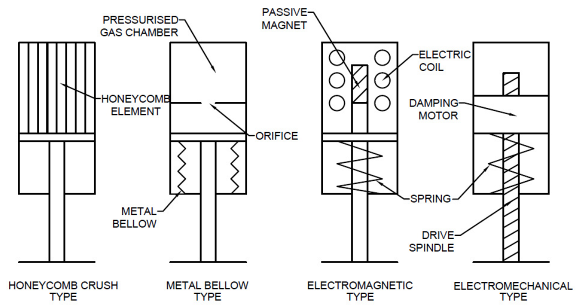

Fig. 19. Simplified representation of various shock absorber systems

Three other possible shock absorbers for space environment which have been previously studied are the metal bellow shock absorbers [33], electromagnetic shock [34] absorbers and electromechanical shock absorbers [35]. Simplified representations of the four systems are shown in Fig. 19.

Metal bellow absorber system uses an orifice between pressurised gas chambers as a passive damping method. The metal bellow and gas chamber both act as a spring element. Metal bellows are designed to operate from cryogenic temperatures to high temperatures. Since the system is gas pressurised and is hermetically sealed, it eliminates the risks of leakage through seals.

Electromagnetic shock absorber uses a passive electromagnetic system as damping method and a spring for resetting method. Since the damping method is achieved through electromagnetic means, no hydraulic or pneumatic system is required. It is easy to make it reusable. However, the usage of magnets and coils add a considerable amount of mass to the shock absorber.

Electromechanical shock absorber has been used in the Philae lander landing system to absorb kinetic energy by driving an electric generator. The electrical energy is thereafter dissipated by resistor. Resettable method can be added by installation of a spring to reset the position of the landing system for reusability. The complexity and mass of the system must also be taken into account while selecting the shock absorber.

Further study is on-going to select and design the absorber to meet the requirements for the preliminary design parameters of the RLRV landing system.

5.2 Structural design

The preliminary structural design of the RLRV has been further refined. Compared to the results presented in [10], several aspects were added to improve the sizing. This is particularly the case of the landing gear. The landing gear design, indeed, influences the mass breakdown through the landing gear itself, but also through the main structural design for which the landing load cases have to be taken into account.

The mass of the landing legs has been determined, according to the method presented in 5.1 for different configurations. In particular the H20d4d3 and H20d3d3 configurations of [10], both containing up to 20 tons of LOx and LH2, and with LH2 and LOx tank diameters of 4 m and 3 m, in one case and of 3 m for both tanks in the other case, were considered for the landing leg sizing. Results from [10] showed that the second and slender configuration has lighter tanks, inter tank and front and rear skirt structures than the other. It is however also characterised by a higher centre of gravity at landing. This has a negative impact on the landing stability and required much larger and thus heavier landing legs, see section 5.1. These results lead to prefer more compact designs such as H20d4d3. However for several reasons it appears that choosing different diameters for both tanks as adverse consequences. Using the same diameter for both tanks is advantageous for manufacturing, for leg folding during launch and refuelling operations, and allows reducing the length of the secondary struts. Consequently a compromise has been found with a common tank diameter leading to a small cylindrical part for the LOx tank such as a H20 d3.5d3.5, i.e. with both LOx and LH2 tanks using a 3.5 m diameter.

The dynamic landing simulations performed and presented in section 5.1, consider different landing load cases. Their influence of the whole structural design has been taken into account. Specifically, landing load cases with no payload, 10 ton and 25 ton payloads have been considered, in addition of the launch, ascent, descent and in-space operation load cases. Additionally both 0° and 2° inclinations of the landing area have been considered. The positions of the interfaces between landing legs and vehicle have been varied as well. They indeed directly influence the structural sizing of components such as the lower skirt and the inter-tank structure.

Based on these data and a refined subsystem mass budget in particular at propellant management level (see section 6), a preliminary structural design has been performed for different tankages between 20 and 120 tons of LOx/LH2 in prevision for the propellant to be transferred to the RTLV in case of LOx and LH2 both produced through ISPP on the Moon. The DLR in-house program 'lsap', based on the Euler-Bernoulli beam theory was used to size the structural elements which are the tanks, the engine thrust frame, the payload adapter cone, the inter-tank structure and the front and rear skirts (see Fig. 20).

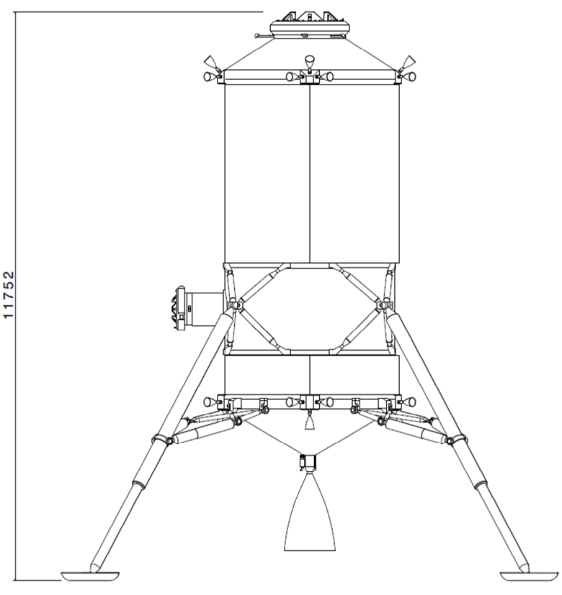

Fig. 20. Sketch of the RLRV for a 20 ton tankage

As already explained in [10], in order to limit heat flux to the propellant tanks, stiffened structures are not favoured. Stringers and frames artificially increase the contact area between structure and propellant, in case of internal stiffeners. In case of external stiffeners the structural surface able to absorb heat from the environment is increased. Luckily the computations showed that for all tankages, while increasing progressively the diameter from 3.5 m to 4.5 m for larger tankage, shell design is the lightest design for the LH2 tanks which are particularly sensitive to heat fluxes. For the LOx tank, the shell design is the lightest for lower propellant loading. For higher and increasing tankage, stiffened tanks allow an increasing mass saving with respect to shell design.

For the unpressurised structures which are the payload adapter cone, the front skirt, the inter-tank structure, the rear skirt and the engine thrust frame, a sandwich structure results in the lightest masses. Note that the landing legs are attached to the lower skirt and the inter-tank structure. During launch the RLRV is connected to the Earth launcher through the rear skirt. While mesh design without skin was considered in the preliminary design presented in [10] for the unpressurised structures, two reasons motivated a change.

First during the mission, micro-meteorites may impact and damage the RLRV. To protect the RLRV from most micro-meteorite impacts, Whipple shields can be used [36]. It consists of a thin bumper placed at some distance of the component wall. While impacting the bumper, the micro-meteorites break-up or even vaporize, leaving only a multitude of much smaller and slower particles to impact the component to be protected. In the case of the RLRV, avionics and piping for instance should be protected and while sandwich structures are not really designed as Whipple shield bumpers, they act as such, up to a certain level. The sandwich structures also protect the dome of the propellant tanks. The cylindrical part of the tanks should be further protected, especially if the RLRV has to be oriented in such way that the tank cylindrical side is becoming the front of the vehicle. As a matter of fact, due to the vehicle velocity adding with the velocity of the micro-meteorites assumed to be homogenously distributed in term of velocity and direction in free space, the front of the vehicle is the most exposed to damage. The RLRV can however be rotated along its longitudinal axis so that always the side opposite to the ADAPT docking and propellant transfer system is in the front regarding the dominant component of the interplanetary dust flux. In such a way only half of the tank circumference needs to be protected by an additional Whipple shield bumper which can also be discontinued at the level of the inter-tank structure, to save mass. Impacts on the non-protected parts are however possible. Further studies should show if the risk level is acceptable

These Whipple shield bumpers and unpressurized structures also have the advantage to protect completely the tanks from direct sunlight and thus reduce the heat flux into the tanks. Only the heat they are themselves radiating and the heat transfer conduction through attachment is incoming in the propellant tanks and can contribute to boil-off.

In the range of propellant tankage from 20 tons to 120 tons, the structural index, defined as the ratio of the dry mass (without engine) divided by the total propellant mass is varying from 16.5% to less than 8%. Note that 10% mass margin has been applied to all components. Further decrease of the structural index could be reached. The structural design has indeed been performed considering a launch from the Earth with full tanks. Launching with only the propellant needed to fly to the Moon and possibly to land a 10-tons payload could strongly decrease the propellant loading at launch. In addition smaller Earth launch vehicle would be sufficient. The structural sizing showed also that the load case corresponding to the launch from Earth has a large impact on the final structural mass. Limiting the propellant mass at launch would decrease the mass of the whole structure and especially of the tanks. Note that in none of the computations, the payload is attached to the RLRV during the launch. This reduces bending loads on the structure.

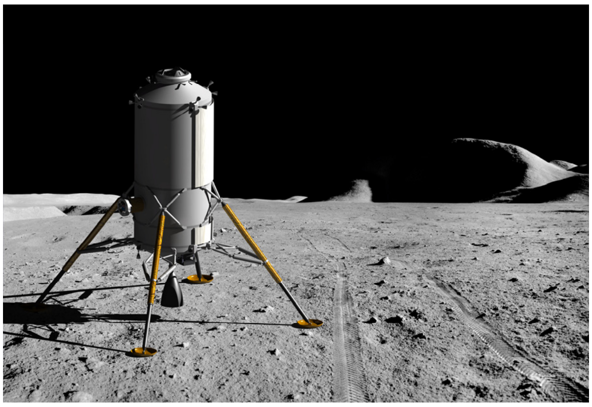
Fig. 21. Artist impression of the RLRV on the Moon

As shown in Fig. 21, the RLRV is landing on classic landing pads. Due to the relatively important height at which the payload is situated a crane would be needed to unload the RLRV. It could be brought during one of the first missions. Displacing the RLRV could be therefore advantageous. Adding wheels to the RLRV could help, but they would have to be carried all the time and with increasing RLRV propellant loadings, the required wheel size could become very large to avoid sinking in the regolith. An attractive solution could be provided by four cooperating rovers docking with and lifting each of the RLRV landing legs to move it from the landing area to an unloading area. This solution will be the subject of further analyses.

6. RTLV

6.1 Preliminary design and mass

As for the RLRV (see section 5.2), a preliminary structural design has been performed for the RTLV, with the goal to derive the structural index as a function of the vehicle tankage. Analyses have been performed with the help of the DLR in house software 'lsap' based on the Euler-Bernoulli beam theory. Load cases considered for the sizing are the launch from Earth, and in space flight load cases with or without payload, with different propellant fuelling level and with engine on or off. Note that the launch from Earth is performed without any payload attached to the RTLV. During space operations, the thrust level, in the range in which the engine can be throttled, is not influencing much the ΔV. Consequently even for a low propellant level and without payload, the acceleration can be kept at moderate levels. Tankages from 20 tons to 120 tons have been considered with tank diameter increasing progressively from 3.5 to 4.5 m. Note that for synergy and costs reasons, it would be preferable to use the same diameter for both tanks of the RTLV and even if possible the same diameter as for the RLRV.

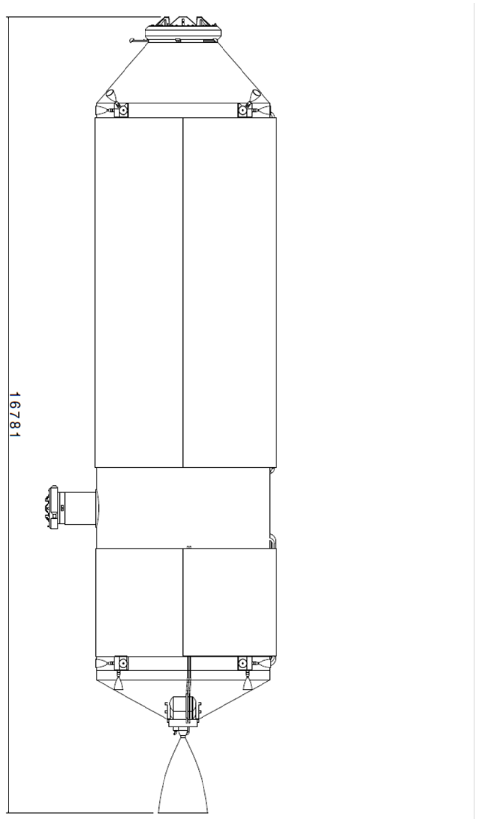

Fig. 22. Sketch of the RTLV for a 40 ton tankage

The general architecture of the RTLV is very similar to the one of the RLRV see Fig. 22. Similarly, a separate tank architecture is selected to limit heat transfer between the LOx and LH2 tanks. For what concerns the loads there are however quite different. In particular neither landing loads nor corresponding bending loads are to be considered for the RTLV. The resulting optimal structure is therefore a bit different. While the engine thrust frame and lower skirt are optimal in sandwich structure as on the RLRV, the other unpressurized structures are getting lighter when stiffened structures are used. The tanks, as for the RLRV (see 5.2), should preferably be manufactured as shell structures, to limit heat flux from the outside. Structural sizing shows that for all propellant tankage between 20 and 120 tanks, shell design is leading to the lowest mass for the LH2 tank. For the LOx tank of tankage getting over 60 tons, a stiffened shell structure should be probably preferred to keep the structural mass as low as possible. Based on these preliminary structural design results, it was computed that the structural index without engine of the RTLV, including a 10% margin is decreasing with increasing propellant mass from about 13.5% for 20 tons tankage to slightly less than 6% for 120 tons. Note as well, that as for the RLRV, the sizing is considering full tanks at launch. Depending on the mission scenario and whether ISPP on the Moon is used for both LOx and LH2, lower fuelling level could be sufficient and the structural index could be further reduced. This effect would be particularly noticeable for the large tankage, as the launch case has a large influence on the tank structural sizing.

6.2 Propellant management and transfer

In the proposed Moon transportation system, both the RLRV and the RTLV are reusable, they should be consequently refuelled. While in the case of ISPP producing LOx and LH2 on the Moon, the RLRV could be refuelled directly on the lunar surface, the RTLV requires in any case an in-space propellant refuelling capability. In case of ISPP producing only LOx, LH2 would have to be transferred in space as well to the RLRV. More generally the cryogenic propellant management for the planned several day missions has a strong impact on the transportation system sizing. Preliminary results are presented in the following subsections.

6.2.1 In-space propellant transfer

In order to perform in-orbit refuelling, it will be necessary to allow in space cryogenic fuel transfer. Due to the specific behaviour of cryogenic fluids in combination with weightlessness, this process is not trivial. In fact, in a cryogenic tank in weightlessness, the liquid and the gases resulting from the liquid boil-off will be floating within the tank. In particular, it cannot be guaranteed that the tank outlet will be permanently in contact with liquid phase. This is problematic as for an efficient fuel pumping, as little gas as possible should be pumped. Several methods exist to guarantee that only liquid will be pumped. A classic solution for in-space systems is to use a large propellant management device (PMD). By capillarity liquid propellant is kept next to the outlet. Another method would be to settle the propellant with a small thrust. According to Kutter et al. [37], an acceleration of 10^{-5} m/s² would be sufficient and could be achieved with relatively small engines. Performing the settling can also ease releasing the excess gas from the tank and by doing so reduces the temperature in the tank which also reduces the boiling off of the cryogenic propellant [37]. The gases that are released are usually just vented out. These gases however still contain energy and can be used to power other systems such as an auxiliary power unit. This idea has been developed by ULA with the integrated vehicle fluid system (IVF) that can use gaseous hydrogen and oxygen from the propellant boil-off, to provide amongst other power and gas to feed RCS thrusters [38].

6.2.2 Auxiliary power unit

In principle, the auxiliary power unit (see left part of Fig. 23) will extract the gaseous oxygen and hydrogen along some of the LOx and LH2 from the tanks. GOx and GH2 can be used to supply a small conventional internal combustion (IC) engine. This engine can drive for instance a generator producing a source of electrical power and by so reducing the mass of battery or/and solar panel. The IC engine will also produce heat and mechanical power. They can be used to turn LOx and LH2 into pressurized GOx and GH2 by driving two small pumps and also by vaporizing the LOx and LH2 to cool down the engine. These pressurized gases can be used for main tank pressurization and replace helium, which cannot be gathered on the Moon. The system can also supply low pressure GOx/GH2 RCS thrusters which allows removing completely any other propellant such has hydrazine from the spacecraft. This system could allow the whole spacecraft to run only on the LOx and LH2 and then take advantage of ISPP.

ULA is currently developing the IVF for its future ACES upper stage, which in some versions should perform Moon missions [39]. The mission characteristics are consequently close to those of the RLRV and the RTLV. The IVF as currently developed by ULA burns about 2 kg/hr of hydrogen and 1 kg/hr of oxygen at low power mode. For power peaks the consumption can increase up to 12.5 kg/hr and 6 kg/hr for H_2 and O_2 respectively [38]. As LH2 is boiling-off much quicker than LOx, a non-stoichiometric combustion is preferred. The starter/generator equipping the IVF is expected to deliver an electrical power of up to 4 kW. This can allow a strong reduction of the battery capacity which would be needed as buffer and energy source to start the IC engine. ULA estimated that the batteries mass could be reduced, in the best case, by 94% on this type of spacecraft [38]. In addition the exhaust from the IC engine could produce enough thrust to create permanent settling. As for the RLRV and for

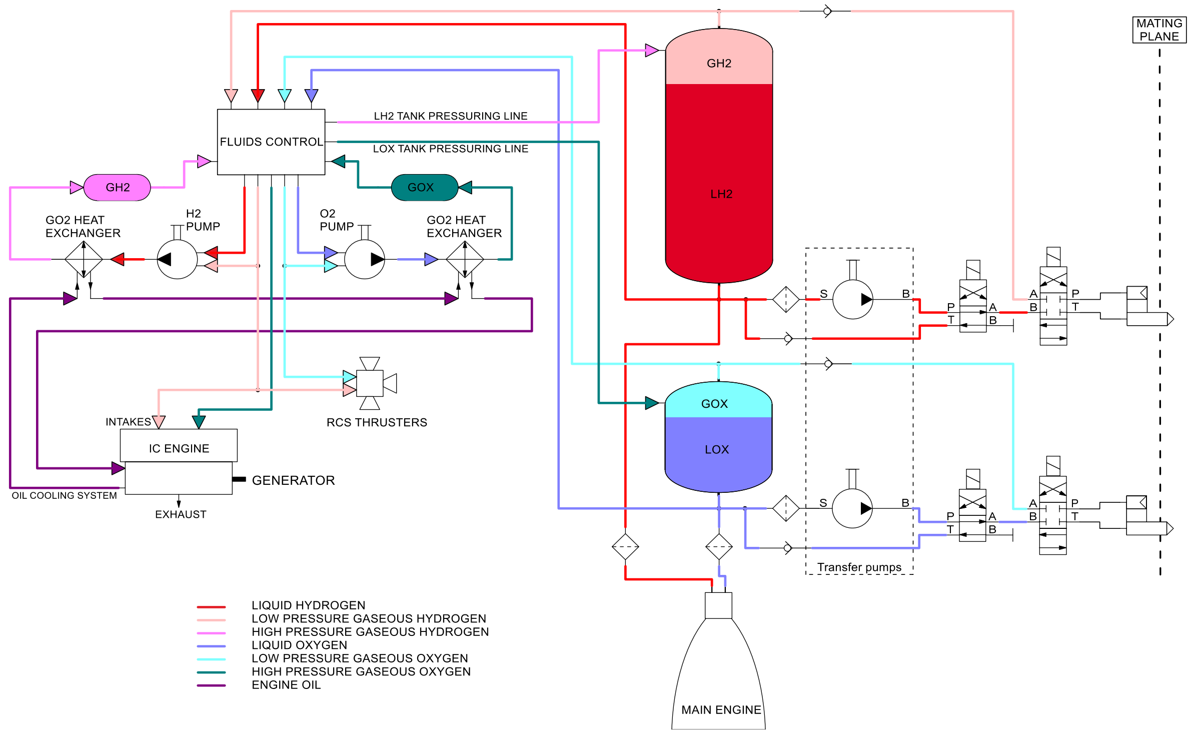

Fig. 23. Overview of the propellant management system of the RLRV and the RTLV. Left of the tanks a system based on ULA's IVF [38] is represented. On the other side the propellant transfer system: ADAPT is displayed.

the RTLV the requirements are not completely the same as on the ACES upper stage, a system similar to the IVF but adapted to the RLRV and RTLV requirement is under study at DLR. Note that to increase synergies between the RTLV and the RLRV, the propellant management system presented in Fig. 23 should be the same for both vehicles.

6.2.3 Propellant transfer system concept

Under the assumption that a low acceleration settling is achieved during the propellant transfer, a propellant transfer system has been the object of a preliminary design.

The schematic in the right part of Fig. 23 presents the propellant transfer system concept ADAPT (Advanced Docking And Propellant Transfer). Propellant transfer should be possible between a RTLV and a RLRV, but also between two RTLVs or two RLRVs to allow collaboration between vehicles. These systems are reversible which means that they can either pump out the propellant or receive it through the same ducts. To be reversible the systems incorporate valves to bypass the pump when the system receives the propellant. The same feedline is used to fill or empty the tank. A second set of valves can close the circuit and stop the propellant flow through the connectors. The same valve can also invert the connectors. Indeed for each propellant two lines are used, in one line the liquid propellant is pump from one tank to the other. The other line allows the gas to constantly equalise the pressure between both tanks. The connector plate design is fully androgynous. Thus two identical spacecraft can dock and transfer fuel without any compatibility problem. This system is sized to allow the spacecraft to simultaneously, if necessary, transfer both LOx and LH2 in order to limit the fuel consumption required for the settling.

6.2.4 Preliminary design of the ADAPT system

In order to perform the propellant transfer, the vehicles have to firmly dock with each other. Consequently the ADAPT system comprises a fully androgynous docking system derived from the International Low Impact Docking System (iLIDS) [40] developed for the International Space Station according to the IDSS (International Docking System Standard) [41]. In the case of the Moon transportation system it is planned to use the exact same docking system on each spacecraft. In such a way they can all collaborate and transfer propellant with each other, making the transportation system more robust and adaptable.

Using the exact same principles as on the iLIDS ensures the androgyny and low impact dockings. For the RTLV and the RLRV neither crew nor equipment will be exchanged through this docking system and only very low thrust levels are intended to be applied while docked. Therefore the diameter has been reduced by 50% compared to the original IDSS. The space within the ring will feature the cryogenic fuel connectors that will be used to perform the fuel transfer. The mating ring features the same hook-based hard locking mechanism as on the IDSS [41] in order to allow the two spacecraft to be firmly attached to each other.

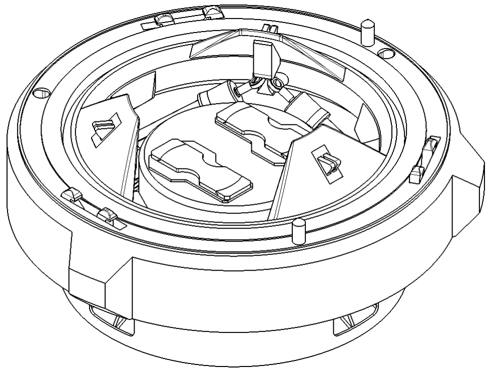

Fig. 24. ADAPT system shown in fully retracted position (CATIA V5 CAD Model)

The ADAPT system is shown in fully retracted position (connector and capture ring) and with the extended cryogenic fuel connector in Fig. 24 and Fig. 25, respectively. The deployable fuel connector is a rather complex system that allows the two spacecraft to connect automatically and repeatedly to each other in order to transfer fuel. Similar types of connectors are used for the Ariane 5 LOx and LH2 fuelling and VACCO performed a mono-propellant transfer test on the Orbital Express mission also with a similar design [42].

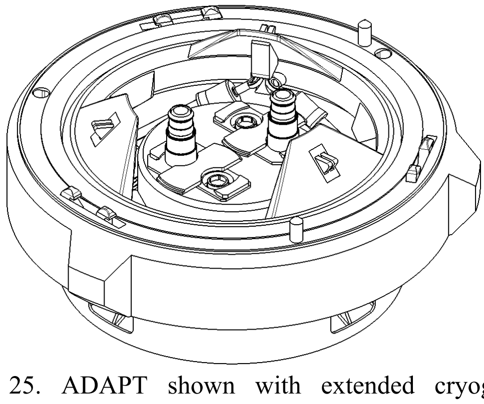

Fig. 25. ADAPT shown with extended cryogenic connectors (CATIA V5 CAD Model)

7. Preliminary comparison of the lunar transportation options

Based on the results presented in the previous sections, in particular the ΔV and TOF given in Table 1 and the structural index evolution, a preliminary design of the Moon transportation system was done for different staging orbits in the Moon vicinity. Results presented here are limited to the case of LOx and LH2 ISPP on the Moon. Additionally to the transfer duration, time has been considered for phasing, docking, propellant transfer and payload transfer from one vehicle to another one. It is also assumed that the auxiliary power unit is constantly running to provide power or any other services that might be required. The selected engine has a combustion chamber pressure of 80 bar. For the RTLV the expansion ratio is set at 250, whereas it is set at 200 for the RLRV. The specific impulse is reduced by 2 s to keep some margins. The engine thrust is adapted to guarantee a thrust to Earth weight ratio of at least 0.5 at launch or beginning of the descent from LLO of the RLRV for the reference mission with a 10 ton payload.

For the sizing, the reference mission considers a 10 tons payload in LEO. This payload has to be landed on the Moon. Three different options are considered:

- Option 1: two RLRVs from the Moon surface are needed to refuel one single RTLV in a cis-lunar orbit. The RTLV performs a boost towards Earth, inserts into LEO and docks with the payload. It then returns to the cis-lunar orbit, where a RLRV takes the payload down to the Moon surface.
- Option 2: two RLRVs from the Moon surface are needed for each of the two RTLVs that will depart towards Earth. Once full, the two RTLVs dock together depart from the cis-lunar orbit and insert into LEO. One of the RTLVs transfers part of its propellant to the other and flies back to the cis-lunar orbit. The other RTLV docks with the payload and takes it to the cis-lunar orbit. From there a RLRV takes the payload down to the Moon surface.
- Option 3: this option is not considering ISPP and the vehicles are not reusable, but the structural index curves for the RTLV and RLRV are reused to size them. As the mass penalty due to the reusability of the components is small, it is neglected in this preliminary estimation. The RTLV and RLRV are renamed ETLV and ELRV, respectively with the "E" standing for expendable instead of the "R" of reusable. The ETLV and the ELRV are launched in LEO. From there the ETLV brings the payload and the ELRV to the Moon surface.

For a given option and a given cis-lunar orbit, all RLRVs are the same and all RTLVs are the same. Their reusability gives them the opportunity to be used for different tasks. The tankage is determined based on the maximum propellant loaded to perform the aforementioned missions with only residuals and reserves left until the next fuelling is performed. In the case of the reusable vehicle they are carrying a payload on their first flight. In option 1 and 2 we have a ratio of two RLRVs for one RLRV. On the first flight of the RTLV it is fuelled with enough propellant to carry one new RLRV fuelled with enough propellant to land. The RLRV is then refuelled on the Moon and bring propellant to the RTLV to bring a completely empty second RLRV. This second RLRV, waiting in LEO, is then refuelled in a cis-lunar orbit with the other RLRV.

Seven different cis-lunar orbits (and transfer options) have been considered for comparison:

- LLO: low lunar orbit
- EML1s: Earth Moon Lagrange point halo orbit with a short transfer time and thus a higher ΔV
- EML1l: Earth Moon Lagrange point halo orbit with a longer transfer time and thus a smaller ΔV
- EML1ld: Earth Moon Lagrange point halo orbit with a longer transfer time and thus a smaller ΔV but with a descent directly towards the Moon surface without stopping in LLO during the descent
- SDRO: 10000 km Selenocentric Distant Retrograde Orbits (only for near equatorial landing sites)
- NROp: near rectilinear orbit for near polar landing sites
- NROe: near rectilinear orbit for near equatorial landing sites

For each cis-lunar orbit, the vehicles needed to perform the missions described for the three options have been sized. The resulting cumulative masses to be launched to LEO are plotted in Fig. 26.

Using EML1 halo with a short transfer time always leads to the highest masses. Flying directly between an EML1 halo and the Moon surface is not advantageous with respect to a similar transfer going through a LLO. Amongst all cis-lunar orbits, LLO staging leads to the lowest mass to be injected to LEO. The advantage over NRO is however small in the case of a transportation system with reusable vehicles. NRO has the advantage to offer a good repartition of the workshare between the RLRV and the RTLV avoiding that one is reaching its engine end of life much quicker than the other. This is not the case when LLO is chosen as a staging orbit. In this case, the RTLV is reaching its end of life two to three times faster than the RLRV. EML1 halo with longer transfer time and SDRO also allow a good share of the work between the RLRV and the RTLV. The mass to be launched in LEO for these two cis-lunar orbits is slightly higher than for NRO.

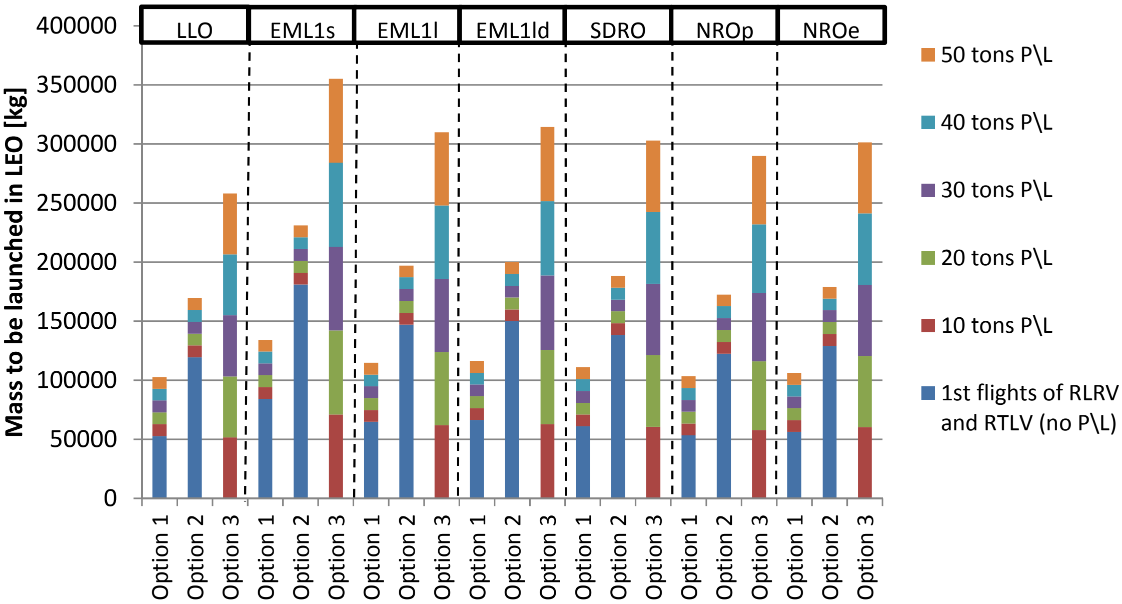

Fig. 26. Cumulative mass to be launched to LEO to land 10 to 50 tons payload on the Moon

When comparing the different options it appears clearly that the expendable case (option 3) displays much higher requirement in term of payload to be launched in LEO.

One assumes that the reusable vehicles can be used for the transportation of five time 10 ton payload. This assumption requires engine life time between 3000 s and 8000 s depending on the cases. The mass to be launched in LEO is 2.5 to 2.8 smaller in option 1 than in option 3. The ratio between option 2 and option 3 is between 1.5 and 1.7. If the number of reuse is increased, these ratios further increase. For instance for NRO with an engine life time of about 4500 s, the RLRV and RTLV can perform five missions. In option 1 about 105 tons have to be launched to LEO to land 50 tons of payload on the Moon. In option 2, about 175 tons have to be launched to LEO and about 300 tons for option 3 for the same 50 tons of payload on the Moon. This means that very quickly reusability linked with ISPP is getting advantageous, even when considering the equipment required producing propellant on the Moon.

For a long term presence on the Moon this would reduce strongly logistics costs. No very large launcher would be needed either. Option 2 appears not competitive compared to option 1. It could however probably be improved with a more elaborate cooperation between the two RTLVs, as the tanker does not have to be injected in LEO. Option 2 also considers smaller vehicles which therefore might allow using a broader range of Earth launch vehicles. Further analyses are on-going in order to consider additional operational scenarios.

Cost saving at logistic level, can compensate the cost of the needed technological development such as those related to ISPP, propellant transfer but also allow performing additional missions. The possibility to refuel vehicle with propellant from the Moon either in LEO or in a cis-lunar orbit also opens new possibilities and maybe even some commercial markets.

8. Conclusions

A Moon transportation concept has been proposed. It is based on reusable vehicle running on turbo-pump fed rocket engines running on LOx and LH2. In-situ propellant production should be used to decrease the amount of payload to be launched into LEO. The transportation system is made out of two different vehicles: a Moon single stage to orbit vehicle also able to land on the Moon called the RLRV and a transporter stage flying between a LEO and a cis-lunar orbit, called the RTLV. Various cis-lunar orbits have been considered. Preliminary models of the RLRV and of the RTLV have been established and in particular the correlation between the propellant tankage and the vehicle dry mass has been determined. This was done with structural sizing and optimisation, taking into account the results of landing dynamic analyses for the RLRV. Based on these results and further subsystem preliminary design analyses, the vehicles have been sized for different cis-lunar staging orbits and different operational modes.

Depending on the selected operational mode, the payload to be launched in LEO can be reduced by at least 2.5 times if reusability is implemented. This value can be increased for extended engine life time.

Performing a staging in LLO lead to a small advantage in term of mass launched in LEO, but other cis-lunar orbit such as NRO, ELM1 halo and SDRO have operational advantages concerning vehicle life time and in some case Moon surface accessibility. NRO, in particular seems promising.

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