67th International Astronautical Congress (IAC), Guadalajara, Mexico, 26-30 September 2016.

IAC-16- D2.3.11

Suitability of Reusability for a Lunar Re-Supply System

Etienne Dumont

Department of Space Launcher Systems Analysis (SART), Institute of Space Systems, DLR, Robert Hooke Straße 7, 28359 Bremen, Germany, etienne.dumont@dlr.de

Abstract

Moon landers built in the past were designed and optimized for specific missions, with a limited consideration of follow-on missions. They were expendable and based on storable hypergolic propellants. In order to enable a sustainable presence on the Moon for research but also other activities, a new type of transportation architecture is needed. Technical developments since the 1960s open a large range of new solutions which could be beneficial for a lunar transportation. The ROBEX (Robotic Exploration under Extreme Conditions) project, in particular, is focusing on new types of lunar architectures, for which modularity, re-configurability and flexibility should play a central role to guarantee the sustainability of the design. In this framework a reusable lunar single stage to orbit vehicle called RLRV (Reusable Lunar Resupply Vehicle) has been designed. It is able to land payloads on the Moon and to launch payloads from the lunar surface in anticipation of their return back to Earth. The RLRV is characterized by a design relying on the combination of cryogenic propulsion, in-situ propellant production and reusability. Important technologies that are enabling such a design have been identified to refine the RLRV design. The flexibility of the vehicle has been demonstrated with the assessment of its performance for a whole range of missions. The comparison of the RLRV with a classic lander such as the descent module of the Apollo Lunar Lander demonstrates that a large reduction of the payload to be injected by an Earth launch vehicle and an Earth departure stage can be achieved with the proposed design.

Keywords: Moon mission, Reusability, ISPP, Cryogenic propellant, ROBEX

Nomenclature

Isp	Vacuum Specific Impulse	s
MR	Engine Mixture Ratio	-
Pcc	Combustion Chamber Pressure	bar
T	Thrust	kN
W	(Earth) Weight	N or kN
ΔV	Velocity Increment	m/s or km/s

Acronyms/Abbreviations

API	Advanced Porous Injector
APU	Auxiliary Power Unit
EML1	Earth Moon Lagrange Point 1
ESAS	Exploration Systems Architecture Study
GH2	Gaseous Hydrogen
GOx	Gaseous Oxygen
ISPP	In-Situ Propellant Production
IVF	Integrated Vehicle Fluids
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLH2	Lunar (produced) Liquid Hydrogen
LLO	Low Lunar Orbit
LLOx	Lunar (produced) Liquid Oxygen
LM	Lunar Module of the Apollo programme
LOx	Liquid Oxygen
LSAM	Constellation Lunar Surface Access Module
NPSP	Net Positive Suction Pressure
NTO	Nitrogen Tetroxide
RCS	Reaction Control System
RLRV	Reusable Lunar Resupply Vehicle
SSTO	Single Stage To Orbit
UDMH	Unsymmetrical Dimethylhydrazine

1. Introduction

In the past, several missions managed to land safely on the Moon and were big scientific successes such as the Surveyor and Luna robotic programs or more recently China's Chang'e 3 mission. These missions together with the Apollo program helped to understand better our natural satellite. However none of these programs succeeded to allow a long term presence and study of the Moon. Such a goal is however currently increasing in interest in particular under the impulse of the "Moon Village" initiative launched by ESA's general director J. Wörner.

Key assets which would greatly help to increase the sustainability of a new Moon program are modularity and reusability. Indeed landers built until now were designed and optimized for very specific missions, with a limited consideration of follow-on missions. They were expendable and based on storable hypergolic propellants. While storable propellants are particularly adapted to several day long missions and the hypergolic property increases the reliability of the mission, it implies the fueling of the vehicle with relatively large mass of propellant coming from the Earth. More recent studies and plans to come back to the Moon such as the ESA Lunar lander or the ESA Cargo Lander also used storable and hypergolic propellants. While these vehicles were already considering a certain modularity and adaptability since they could transport various payloads, they had an additional limitation as their design is constrained by the capabilities of existing launchers available: such as Soyuz or Ariane 5 [1]. NASA with its Constellation program considered the design a bit differently, as from the very beginning the payloads to be launched and the corresponding lander (Altair) were designed with limited constraints on the capabilities of the launcher. As a matter of fact for this program a new launch vehicle, Ares V, in the class of Saturn V was planned to be built. However, with such architecture, regular missions to the Moon imply the production and the launch of huge and expensive launch vehicles. It cannot be sustained easily by any space agency in the world.

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 in the past years. These developments are indeed allowing considering as reasonably feasible, designs, which for instance in the 1960s were considered as far too risky. This is the case for instance of cryogenic propulsion for which experience has been gathered for already many decades [2]. In-Situ Propellant Production (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 [3], [4] and [5], and some of them have been tested on Earth [6]. A source of propellant directly on the Moon would allow reducing strongly the mass to be launched from the Earth. Recent studies [7] also indicate that water ice is present on the Moon; this would allow producing not only LOx but also LH2. While these two technologies are expected to improve the performance of transportation between the Earth and the Moon, their combination with reusability seems particularly beneficial. It is indeed expected that a further reduction of the size of the transportation system as well as an increase of the mission flexibility can be obtained. Actually, reusability is expected to be beneficial for the different sections of the transportation system: the Earth launch vehicle, the Earth-Moon transfer vehicle and the Lunar Lander.

In the frame of the ROBEX (Robotic Exploration under Extreme Conditions) project, a new transportation system for lunar missions is being studied. An important aspect of the ROBEX project is the consideration of modularity, flexibility and re-configurability for the design of the ground infrastructure in order to increase its sustainability. This requires performing missions to establish and re-supply the lunar infrastructures such as those described in [3] but also to launch payload back to Earth. This characteristic naturally influences the choice of the architecture and design of the transportation system, which should also be flexible. A lunar lander able, for its reference missions, to land payload masses of 10 metric tons and to re-supply a Moon base has been pre-designed. This vehicle should also have the capability to launch payloads from lunar surface in anticipation of their return back to Earth. For these reasons, we pre-designed a reusable lunar SSTO (Single Stage To Orbit) vehicle called RLRV (Reusable Lunar Resupply Vehicle). This vehicle, which should work as a shuttle between the Moon orbit and the Moon surface, should have the capability to be refueled both in Moon orbit and on the Moon surface. It has been chosen to consider liquid oxygen and liquid hydrogen as propellant for the RLRV.

After elaborating on how newly developed technologies and researches allow considering the combination of LOx/LH2 propulsion, ISPP and reusability to design an efficient and flexible transportation system, an overview of the architecture, focusing on trajectories, staging and the origin of LH2 is proposed. A preliminary sizing of the RLRV including for instance the design of the structure and of the feed and propulsion systems will be presented. Missions and performance are following. In order to assess the advantages of the reusability of the RLRV, it is compared to the Apollo Lunar Module, a classic expendable vehicle. For that purpose, different mission scenarios amongst which the establishment of the lunar base with habitat, pre-sized previously in the frame of ROBEX [3] are considered.

2. Design rationale

With the exception of China's Chang'e 3, all soft landings on the Moon occurred between 1966 and 1976. These 19 missions took place in the frame of the Space Race between the Soviet Union and the United States of America. The main design driver was at that time to be the first one on the Moon while sustainability or cost of a design was only a secondary aspect. From this period a lot of experience has been gathered and still strongly influences the design of vehicles for future missions. Basically, these designs can be summarized as being based on pressure-fed engines running on terrestrial based storable propellants, often relying on Low Lunar Orbit (LLO) for rendezvous or at least as a parking orbit and being expendable. Several studies performed in the past years came out with ideas, which could bring a cost reduction in comparison with Apollo-type designs, qualified by Zubrin as "brute-force means" [5]. But as regretted by Zubrin, these ideas such as In-Situ Propellant Production (ISPP) are often applied to classic designs, avoiding benefiting fully from their advantages.

In the frame of this study four 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 engine
- the use of ISPP
- the possibility to replace LLO rendezvous by Earth Moon Lagrange point 1 (EML1) rendezvous
- the implementation of reusability

2.1 Former Moon missions main characteristics

Moon missions, until now, relied mainly on pressure-fed engines running on storable propellants, as it was the case for the Apollo Lunar Module. Sometimes other systems have been combined with the storable propellant pressure-fed engine, such as airbags on Luna 9, the first spacecraft to achieve a soft landing on the Moon, or a solid rocket motor for the Surveyor missions, which helped preparing the Apollo program.

For Moon landing missions, several reasons can explain the choice of storable propellant associated with a pressure-fed engine cycle. First, the simplicity [4]: pressure-fed engines, thanks to very little moving parts, are simple and reach a high level of reliability. The propellant used: mostly hydrazine derivatives and nitrogen tetroxide derivatives are hypergolic, so that no ignition system is required. Second, the mass: this propellant combination has also a high density (about 1442 kg/m³ for NTO and 791 kg/m³ for UDMH), leading to moderate tank volumes. Consequently, the mass penalty of tanks able to bear several bars of pressure, required for pressure-fed engines, is far lower than for less dense propellants such as liquid methane (about 422 kg/m³) or liquid hydrogen (about 71 kg/m³) [8]. Missions to the Moon and back last several days. This is much longer than the missions performed until now with cryogenic propellants, which are not exceeding much more than half a day [2]. Choosing storable propellants simplifies the design of vehicles and their thermal control system. Finally, all missions performed until now disregard the production of propellant in-situ.

Bringing everything from the Earth is indeed easier and might be acceptable when only few missions are planned, but it requires huge launch vehicles such as Saturn V. Of the 16.4 tons at lift-off, less than one third (4800 kg) of the Apollo Lunar Module was dry mass [9]. This is only about 0.16% of the lift-off mass of Saturn V. When considering such a payload ratio, it becomes clear that if this payload ratio could be increased it would make lunar transportation cheaper and therefore more sustainable.

2.2 Propellant and engine for future missions

In order to increase the payload ratio, several studies considered different technical choices with higher performance engines. Plans for new Moon missions in the United States of America considered, for instance, the use of cryogenic propellants (LOx/LH2) for descent stages, amongst other Centaur stage derivatives [10] and [2]. Even the ESAS (NASA's Exploration Systems Architecture Study) baseline for the Constellation Lunar Surface Access Module (LSAM) considered a descent stage running on LOx and LH2 [11]. In all cases RL-10 expander cycle engine derivatives are considered to propel the descent stages. Whereas technical difficulties appear in order to keep the propellant at cryogenic conditions during the mission without allowing too much boil-off and ensuring an acceptable level of reliability, the experience gathered during years of operations of cryogenic stages across the world gives a good confidence that such a technical solution could be adapted to a lander for several day long missions [10], [2] and [12]. If an expander cycle engine burning a mixture of LOx and LH2 is used, one can expect specific impulse in the range of 450 s, which has to be compared with only 311 s in the case of the high expansion ratio descent and ascent engines used during the Apollo missions. Even if the propellant mixture is less dense, it has not to be stored at high pressure and the mass benefit is getting important. For instance, for a ΔV of 1900 m/s, which is typical for a descent from a LLO to the Moon surface, the required mass of propellant with the Apollo Lunar Lander engine is 60% higher than with a 450 s Isp LOx/LH2 expander engine. This propellant mass requirement is even increasing with the ΔV. For a ΔV of 2500 m/s which is typical for a descent from EML1 to the Moon surface the propellant mass surplus for an Apollo Lunar Lander engine is increasing to almost 67%. Relying on expander cycle engines running on LOx and LH2 can, therefore, relax strongly the requirements on the launch vehicle payload capabilities.

2.3 ISPP (In-Situ Propellant Production)

Another possibility to further improve the payload ratio would be to rely on propellant production directly on the Moon, as proposed by Zubrin [5]. In such a way, part of the propellant would not need to be launched from Earth. Unfortunately, not all types of propellants can be gathered on the Moon. According to Hopkins [4], propellant combinations which could be combined with ISPP are LOx/LH2, LOx/silane, and LOx/carbon-based fuels. Probably the best solution would be to go for LOx/LH2, due to a long heritage, the high specific impulse and the relatively high LOx to LH2 mass ratio. As a matter of fact, oxygen can be gathered from Moon regolith almost everywhere on the lunar surface through the reduction of ilmenite [3] or olivine [4] or the carbo-thermal reduction of regolith [5]. Hydrogen and carbon are much more difficult to find but are left by solar winds in the upper layer of lunar regolith in low quantities. Some regions however exist where large quantities of hydrogen can be found. According to [7], one estimates that at least 6.6 x 10^{12} kg water ice are present on the Moon and especially close to the poles. In the case this water ice could be gathered, it would be a very good source of hydrogen through water electrolysis. It could also help to synthesise carbon-based propellants such as methane, methanol or propane, which could be an interesting alternative in the case Mars missions should be flown with similar vehicles. If hydrogen cannot be gathered from the Moon surface, it can be brought from Earth. In this case the propellant mass carried from Earth is strongly reduced, compared to a design relying only on terrestrial propellants. For instance, for a LOx/LH2 system with a mixture ratio of 6, the LH2 mass is just over 14% of the total propellant mass.

It appears first that relying on ISRU (In-Situ Resource Utilization) and more exactly ISPP (In-Situ Propellant Production) could allow reducing the propellant mass brought from Earth for the return leg from the Moon to the Earth. Of course it would be advantageous to take advantage, as much as possible, from the propellant gathered from the Moon. The gravity well of the Moon is as a matter of fact far less deep than the one of the Earth. In average, the ΔV required from the Earth surface to a 400 km Low Earth Orbit (LEO) is between 9.5 km/s and 10 km/s. Table 1 has been established based on trajectory analyses for Moon take-off and landing and Hohmann transfers for the other orbital changes.

Table 1. Velocity increments between Earth and Moon surfaces and LEO (400 km), EML1 and LLO (100 km)

	ΔV between Earth surface [km/s]	ΔV between Moon surface [km/s]
and LEO	9.5	6.3
and EML1	13.3	2.5
and LLO	13.8	1.9

As shown in this table the velocity increment needed to reach a Low Lunar orbit (LLO) or the Earth Moon Lagrange point 1 (EML1) from the Moon's surface is much lower than from the Earth surface. Even if LEO is considered, it is much less energetic to reach it from the Moon's surface than from the Earth's surface. As a consequence, transporting propellant produced on the Moon to LLO, EML1 or maybe even to LEO could help reducing the size of the vehicle needed to fly to the Moon and back, and at the same time the costs. Propellant from the Moon is actually a solution proposed by ULA to reduce the cost of space transportation. However, for ULA [13], Moon missions are only a small part of their so called CisLunar-1000 Vision. According to ULA, the cost of propellant produced on the Moon per kilogram, even in LEO, would be lower than if it is coming from the Earth. Of course such a result depends on many parameters which are function of the assumptions considered in term of cost of the ISPP plant, its reliability and the cost of transportation of propellant from the Earth and from the Moon. In this last aspect, improvements are expected with the introduction of reusability for Earth-based launch vehicle.

2.4 Reusability

Reusability is currently gaining in interest for launch vehicles starting from the surface of the Earth. Blue Origin already demonstrated it with its New Shepard rocket and the launch of a returned first stage of Falcon 9 is expected in the coming months. It is indeed expected that cost reduction can be achieved through the implementation of reusability.

Reusability can be applied to the launch vehicle launching the elements needed for the Moon mission from the Earth's surface but also to the other elements of the transportation chain. This would be possible for both a transfer vehicle flying between a LEO and LLO or EML1 and for a lunar shuttle combining both a lunar lander and an ascent vehicle. Compared to Earth-based systems, the absence of atmosphere strongly simplifies part of the design as no thermal protection system is needed. In addition, velocity increments are quite small. The total ΔV for a round trip between the Moon's surface and a LLO is 3.8 km/s. This is in the same order of magnitude but still less than what has already been achieved for instance by the first stage of Falcon 9 Full Thrust [14]. Flying to EML1 and back would require about 5 km/s.

In the case of a Moon mission, the reusable combined lander and ascent vehicle, named RLRV (Reusable Lunar Resupply Vehicle) [3] would be launched from Earth with sufficient propellant to land on the Moon. Once on the Moon, it could be refuelled with propellant produced in-situ. If only oxygen is produced, sufficient hydrogen should be carried from the Earth for the next ascent. Once again in LLO, or at the EML1, the RLRV would dock with a payload and optionally refuel its LH2 tank, before performing a second descent and a soft landing. Sufficient LLOx (and optionally LLH2) would have been fuelled prior to the ascent to perform the descent with the payload. The RLRV would then be refuelled again and would be ready for its next mission. In such a way the dry mass of a lander would not have to be carried for each mission from the Earth. Moreover considering the low level of thermal and structural loads encountered by such a vehicle compared to a vehicle re-entering the Earth atmosphere, reusability even without maintenance between the missions should be achievable. Actually, reusability for Moon missions has already been proposed for instance by Axdahl [15], however with propellant depots in LEO and supplied from the Earth. With such a method Axdahl estimated that a cost reduction of up to 30% could be achieved compared to the ESAS architecture.

3. Proposed transportation architecture

The proposed transportation system architecture is based on the findings listed in the previous subsections (2.1, 2.2, 2.3 and 2.4). The central characteristics of the proposed architecture are reusability of the vehicles and in particular of the combined lunar lander and lunar ascent vehicle: the RLRV, in-situ resource utilisation to produce LLOx and optionally LLH2 and the use of a turbo-pump fed engine burning LOx and LH2. All these characteristics taken individually are expected to bring a reduction of the cost of Moon missions. The goal, here, is to combine them, to reduce even further the costs. Once that has been set, parameters, which will influence the gain over traditional architectures, can still be varied. These main parameters are:

- the choice of the position for rendezvous between the different elements of the transportation system. Possible positions are LEO and LLO as traditionally proposed but also EML1
- the share of the velocity increments between the transfer vehicle and the RLRV (mainly depending on the previous point)
- the origin of LH2: either gather on the Moon or carried from Earth

3.1 Trajectories and staging

Recalling the velocity increments presented in Table 1, flying directly without staging between the Moon surface and a LEO would require a velocity increment of about 6.3 km/s. Note that in the case of the Apollo mission the return from the Moon to the Earth was less demanding as the capsule directly re-entered in the atmosphere after performing a rendezvous in LLO. Aero-capture, indeed, allows reducing strongly the ΔV for a return to Earth but is not available for the flight from LEO to the Moon surface. A one stage design utilizing aero-capture would correspond to a direct trajectory, as proposed in [5]. While this design might be well adapted to expendable vehicles, it however implies to carry a heat shield all the way to the lunar surface and back and makes the implementation of reusability much more complex. With the type of engine considered for the RLRV (see subsection 4.1) a specific impulse of 452.5 s is expected. Under such conditions the rocket equation tells us that a ΔV of 6.3 km/s can be reached only if the propellant mass is about 76% of the start mass. Considering the fact that payloads of up to 10 tons should be landed on the Moon and that in a reusable case relying on ISPP, propellant for the return branch towards the Moon should be saved, a one stage design would lead to a huge vehicle, outside of the range of what is realistic. Consequently an intermediate position should be selected for the staging between a RLRV and a transfer vehicle. This position should be stable in order to ease reusability. In the case of past missions, LLO has often been selected. Such a rendezvous position has the advantage to limit the ΔV required for the descent to the Moon surface and for the ascent on the return leg. The selected LLO, however, influences strongly, which positions can easily be reached on the Moon. A list mainly based on landing sites proposed for ESAS [16] is given in Table 2. In the case of an expendable system such as the one design for ESAS, reaching these different landing sites can be managed by tweaking the Trans-Lunar Injection (TLI) boost and launching in the appropriate Earth Moon configuration. However for a reusable vehicle, it may also imply in particular cases to perform inclination changes of the LLO between missions. LLO are also not very stable and may require important amounts of propellant for station keeping. The utilisation of the Lagrange points L1 and L2 gained in interest over the last years and they have been proposed already for different missions [17], [13] and [18]. They have indeed some advantages over LLO. As stated by Bobskill [16], these points can be reached and allow access to any points on the Earth as well as on the Moon with limited constraints in term of launch windows and ΔVs. Station keeping requirements are relatively low and space debris are less dangerous as slower than deeper in the gravity well of the Moon or of the Earth. Finally from each of the Lagrange point L1 or L2 about one hemisphere of the Moon can be observed. This is however always the same hemisphere. In term of ΔV, reaching the Earth Moon Lagrange points 1 (EML1) or 2 (EML2) is very similar. However flying to EML2 from the Earth takes much more time. This is an important drawback for cryogenic systems. In the following only LLO and EML1 are considered.

Table 2. Proposed landing sites [19] and [3]

Landing Site	Latitude	Longitude
South Pole	89.9 S	180 W
Far side SPA floor	54 S	162 W
Orientale basin floor	19 S	88 W
Oceanus Procellarum	3 S	43 W
Mare Smythii	2.5 N	86.5 E
W/NW Tranquilitatis	8 N	21 E
Rima Bode	13 N	3.9 W
Aristarchus plateau	26 N	49 W
Central far side highlands	26 N	178 E
Tsiolkovskiy crater	20 S	129 E
North Pole	89.5 N	91 E

3.2 Origin of the LH2

As explained previously in the subsection 2.3, several types of ISPP can be implemented. The two most promising types are relying on regolith from which only LOx can be produced and on water ice from which both LOx and LH2 can be produced. While oxygen rich regolith can be found almost everywhere on the Moon surface, water ice is present mainly near the poles.

Consequently, developing the capability to produce LLOx from regolith would allow using ISPP at each of the landing sites listed in Table 2. The drawback would be that LH2, admittedly in relatively small quantity, would have to be transported from the Earth and boil-off would occur during the relatively long flight.

However, there is a solution to use ISPP based on lunar water ice. It is possible to uncouple the location of the ISPP plant from the main Moon base. Deploying one or several ISPP plants close to the poles would allow producing both LLOx and LLH2. This propellant should then be either transported to the main base or the RLRV should come to an ISPP plant to be refuelled. In both cases this can be achieved through a suborbital flight between the ISPP plant and the main base for a lower energy level than an injection to orbit. While such flights would consume part of the life expectancy of the RLRV, having an ISPP plant close to each pole would allow reaching one of them with a range not larger than one quarter of the circumference of the Moon from any point of the lunar surface. This flexibility could get particularly interesting if several bases are being built, as one or two polar ISPP plants could supply all of them.

4. RLRV design

4.1 Engine

As mentioned in subsection 2.2, current mission designs in the USA tend to favour the use of the RL-10 engine for Moon landers. The RL-10 engine has already a long heritage and has been produced in different versions. Of interest is especially the RL-10 B2 currently in use on the Delta launch vehicle. This expander cycle engine burning LOX and LH2 is able to deliver a vacuum specific impulse of 465.5 s and has a life expectancy of at least 2000 s [17], Axdahl [15] even states 3500 s. In addition, this engine is able to be throttle in the range 8% to 104%, which is an important asset for a lander engine. It is also able to re-ignite 15 times.

A new engine has been the object of a preliminary sizing, and its characteristics have been selected to be well adapted to the RLRV. Compared to the first version of the RLRV presented in [3], both the engine cycle and thrust level have been modified. In the thrust range of interest expander cycle engine are perfectly adapted and they provide better performances than gas generator cycle engines. The thrust level has been also modified for three main reasons.

First, the required ΔV to land on the Moon or to depart from the Moon depends on the gravity losses which themselves are linked to the thrust to weight ratio of the vehicle. Trajectory analyses for ascent and descent of the RLRV with different thrust levels have been performed. The trajectories have been optimised and the part of the mission at full thrust has been maximized. Engine throttling prior to landing is performed only in the very last part of the trajectory, since making this part longer would lead to a higher gravity loss. Thrust to (Earth) weight ratio between 0.15 and 0.64 at engine ignition have been considered. Note that if the Lunar weight is considered, the thrust to weight ratio has been varied between 0.92 and 3.9. The lowest level is only feasible for a descent. In the following, weight will always refer to Earth weight except if differently indicated. The results of the performed analysis are shown in Fig. 1. As expected, the ΔV is increasing with decreasing thrust to weight ratio. This phenomenon is even stronger for ascents than for descents. For ascents the thrust to weight ratio is minimum at lift-off, as the tanks are still full and the flight path angle is close to 90°. For low thrust to weight ratios the RLRV must first fly quite vertical before it gets light enough to accelerate and reach orbital velocity. This can be observed for the T/W range below 0.35. For thrust to weight ratios higher than 0.35 the curve flattens, so that selecting a larger engine has only a small influence on the ΔV, independently of the target/origin: LLO or EML1.

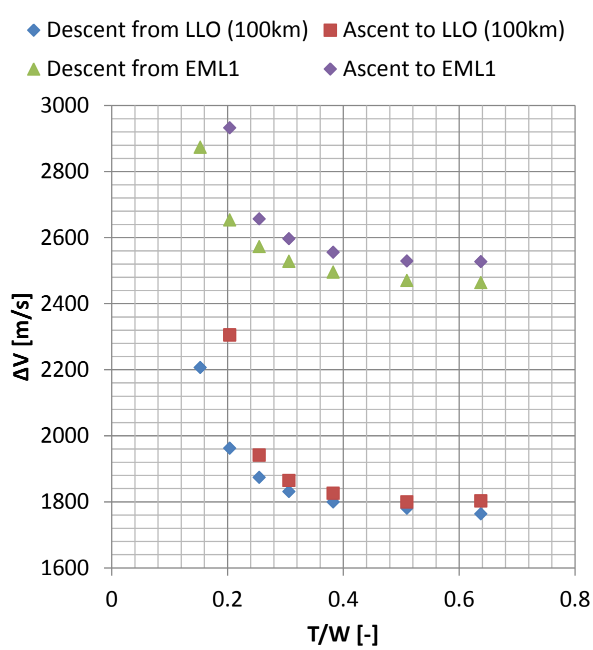

Fig. 1. Variation of the velocity increment for Moon-based vehicle as a function of the thrust to (Earth) weight ratio at mission start

Second, in the preliminary design presented in [3], the engine was derived from the very reliable HM-7B design and the goal was to avoid continuous throttling capabilities, as this is not available in Europe yet. While soft landing without throttling capability works theoretically, not much degree of freedom to perform trajectory corrections is left. For this reason, a relatively low thrust level implying a large ΔV was selected. With throttling capability, this is different, and larger thrust levels can be selected while allowing hovering at landing if needed. Recent developments at DLR have demonstrated that throttleability combined with low production cost and good level of efficiency over a broad thrust range can be reached with the API (advanced porous injector) technology [20]. Deeken even mentions that the API technology allows a decrease in the pressure drop through the injection plate and an increase of the heat flux to the cooling circuit which are particularly advantageous for an expander cycle engine.

The third and last reason why an increase of the thrust level is beneficial is the duration of the burns. Performing an ascent or a descent for a given vehicle is basically the same as burning a certain mass of propellant to generate a velocity increment. For a given Isp, the duration of the burns depends on the ΔV and the propellant mass flow rate of the engine. This mass flow rate is directly linked to the thrust level of the engine. Consequently, reducing the burn time of the engine to complete the mission can be achieved by increasing the thrust level. This point is particularly important in the case of a reusable Moon shuttle, as the engine will be likely one of the most limiting components for the vehicle life expectancy and cannot be serviced on the Moon. Varying the start thrust to weight ratio from 0.2 to 0.64 allows dividing the burn time by 3 for the descent and even about 4 for the ascent, see Fig. 2. This can be directly converted in three to four times as many missions for a given engine life expectancy and providing that the engine can be re-ignited sufficiently often. The reliable re-ignitability of the engine is also an important characteristic of a reusable system. Laser ignition could be the solution to this problem. A re-ignition probability of 100% has already been demonstrated by Börner et al. [21] for a LOx/GH2 mixture during experiments.

The preliminary sizing of seven expander cycle engines with thrust levels between 30 and 125 kN has been performed. They all have common characteristics listed in Table 3. The combustion chamber pressure has been set at 60 bars as for the Vinci engine. A relatively high mixture ratio of 6 for an upper stage LOx/LH2 engine has been chosen in order to limit the amount of LH2 required. It allows limiting the size of the LH2 tank and is advantageous in the case LH2 cannot be produced by ISPP on the Moon. The nozzle expansion ratio has been limited to 100 to limit the length of the nozzle and therefore the height of the RLRV. In addition it is assumed that the life expectancy and throttling range are at least as good as what has been demonstrated by the RL-10 engine. Improvements might be reached with an appropriate design. The resulting engine mass as a function of the nominal vacuum thrust as estimated by the preliminary engine model is plotted in Fig. 3.

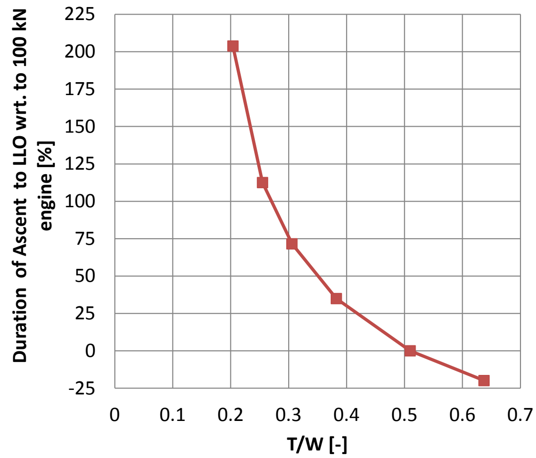

Fig. 2. Duration of an ascent to LLO for different thrust to Earth weight ratio

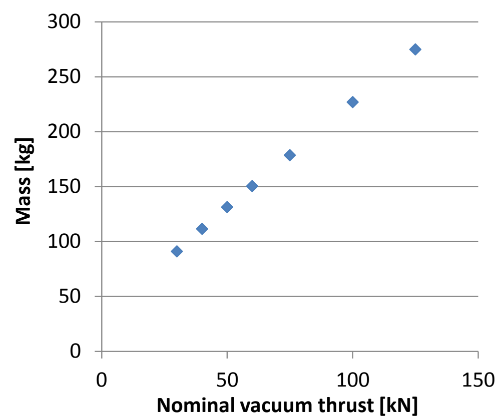

Fig. 3. Variation of the expander cycle engine mass as a function of the vacuum thrust

Table 3. Main characteristics of the LOx/LH2 expander engine

	Value
Pcc [bar]	60
MR [-]	6
Expansion ratio [-]	100
Nominal thrust [kN]	See Fig. 3
Vacuum Isp [s]	452.5
Nozzle exit pressure [bar]	0.039
Mass [kg]	See Fig. 3
Life expectancy [s]	≥ 3500
Throttling range [%]	8 to100

Based on the computed engine mass, the thrust level has been optimised by maximising the payload mass performance. For that purpose, four different structural index laws (excluding the engine mass) have been considered. Among them three are chosen constant: 15%, 20% and 25%. The last one is derived from built LOx/LH2 vehicles propelled by turbo-pump fed engines and thus varying with the propellant loading. The start mass of the RLRV (including payload) has been considered to be 20 tons and the tanks have been sized to carry twice as much propellant as the propellant requirement determined during the trajectory analysis in order to consider an ascent, followed by a descent both at a start mass of 20 tons. 20 tons correspond approximately to the expected mass with a payload of around 7 to 10 tons [3] depending if the rendezvous is performed in EML1 or LLO. The results are plotted in Fig. 4 for an ascent to LLO. Results for a descent from LLO and an ascent to or descent from EML1 are very similar. In each case, the optimum is reached for a thrust to weight ratio situated between 0.5 and 0.6, with a curve already showing losses lower than 2% from 0.3 and lower than 1% from about 0.35 with only a small influence from the structural index law. In the case the engine is getting larger, the payload capability is again decreasing due to the increasing dry mass of the engine which cannot be compensated by the gain in ΔV.

Due to these results, a thrust to weight ratio of 0.5 has been selected. While a thrust to weight ratio of 0.4 would still bring a good performance, the burn time would be significantly increased. For thrust to weight ratios higher than 0.5, very deep throttling would be needed. Consequently, a thrust level of 100 kN has been selected.

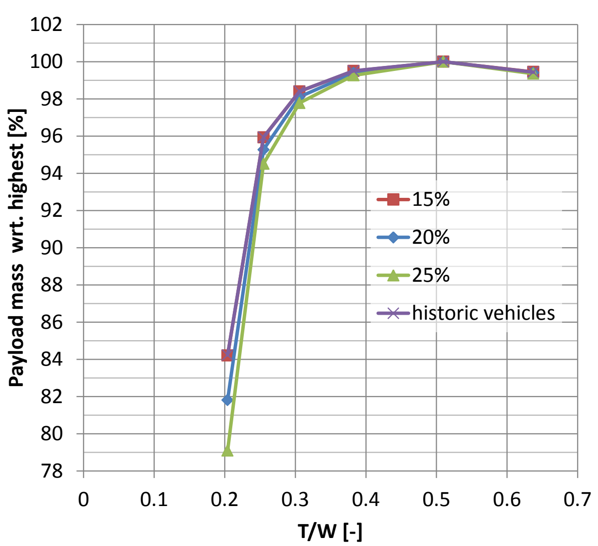

Fig. 4. Payload performance variation as a function of the thrust to weight ratio and the structural index for an ascent to LLO

4.2 Feed system and Reaction Control System

Feed system and Reaction Control system (RCS) design usually rely on consumables such as helium for pressurisation and hydrazine based propellants for pressure-fed thrusters. In such a design the consumables have to be stored in dedicated tanks, adding dry mass to the system. As mission requirements may vary, kits can be installed to increase the quantity of consumables and the capabilities of the stage. This is however linked with an increase of the complexity and of the dry mass of the vehicle. In the case of a reusable vehicle for Moon missions such as the RLRV, adopting this type of pressurisation system and RCS would either limit the lifetime and capabilities of the vehicle or imply a high dry mass. The refuelling of high pressure helium tanks or RCS hydrazine-based propellant tanks might be possible but complex while increasing the dependency on resupply from the Earth. Systems are however currently been developed which could largely mitigate these difficulties. ULA is indeed developing a system named the Integrated Vehicle Fluids (IVF) system [22]. This system is actually using free boil-off hydrogen and oxygen to run an auxiliary power unit (APU). In such a way electricity can be generated, eliminating the need for batteries. GOx and GH2 can be produced for pressurisation even when the main engine is not working. This is important for instance prior to a re-ignition of the engine. It can also provide GH2 and GOx for low pressure RCS thrusters still able to reach Isp in the range of 400 s. Work is also being performed in Europe on laser ignited, low pressure LOx/GH2, 400 N thrusters with a combustion pressure of 2 bar. They could be fed directly from the main tanks [23]. Using these technologies on the RLRV allows simultaneously limiting the dry mass of the vehicle (helium and hydrazine tanks are not needed and batteries can be eliminated or at least reduced), using boil-off propellant which would be lost in other cases and extending the lifetime of the pressurisation system and RCS each time the main tanks are re-fuelled.

A preliminary design of the feed and pressurisation system of the RLRV has been performed. It appears that a tank pressure of 3 bar is well adapted for both the LOx and the LH2 tanks in order to guarantee a sufficient NPSP during the whole mission. Feed lines, pressurisation lines and fill and drain lines have been pre-sized. The mass of pressurant in the tanks along the flight has been computed for different tank designs. First estimations show that 24 low pressure LOX/GH2 400 N thrusters should be sufficient for the RCS of the RLRV.

4.3 Structural design and mass estimation

Landers such as the LM from the Apollo program or the Constellation Lunar Surface Access Module (LSAM) adopted a design where each of the propellants is stored in several tanks, in general two per propellant. These tanks are then placed symmetrically with respect to the longitudinal axis of the lander. Such architecture allows freeing the centre of the lander to place for instance the main engine and limit the height of the vehicle. A limited height indeed eases to unload the payload which is often placed on the top of the lander. However, it has a massive drawback, as it results in a poor structural index, leading to a large increase of the mass to be launched from Earth. This drawback is even increasing when selecting propellants with a low density such as LH2. Sharing the propellant in several tanks has also another drawback, which is specific to cryogenic propellant. The surface to volume ratio of a tank is getting smaller when tanks are getting larger. In the case of cryogenic propellant for long missions, limiting the heat coming into the tank is a major goal to avoid losing too much propellant through boil-off. As a consequence it is much more optimal to store each propellant in only one tank for each. To avoid heat transfer from one tank to the other, a separated tank design is also advantageous. Indeed while LH2 is stored at 21 K, LOx is stored at 90 K. A common bulkhead design would make the thermal management much trickier. All configurations considered in this study focussed, therefore, on a two separated tank design.

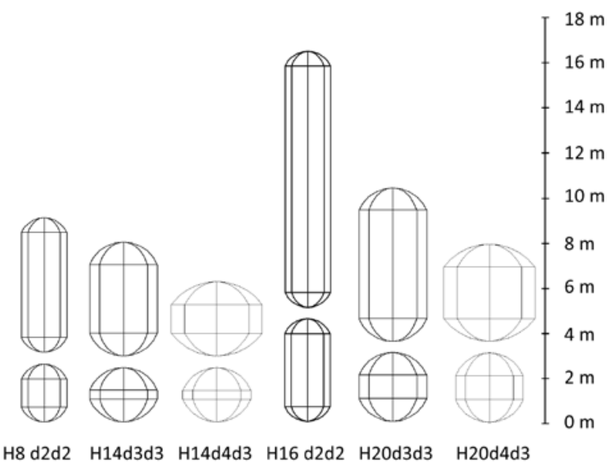

Fig. 5. Selection of tanks considered during the preliminary sizing

As stressed by Birckenstaedt [2], constructive decisions can further ease the cryogenic fluid thermal management. Each internal protrusion increases the internal surface of the tank and consequently eases heat transfer. Consequently, a smooth tank structure is preferred. For the tanks only shell structure design has been selected. In total 18 different tank designs have been considered with propellant loading varying from 8 to 20 tons and diameter varying from 2 to 4 m. For some configurations, the diameters of the LOx tank and the LH2 tank have been chosen different in order to avoid that the LH2 tank is getting to long. Some examples of tank design are compared in Fig. 5. Each tank design is designated under the form *Hx dydz*, where *x* is the propellant loading in tons, *y* is the diameter of the LH2 tanks situated on top and *z* the diameter of the LOx tank both in meter.

Following the tank sizing, a preliminary structural design has been performed. For each of the 18 tank designs, 6 load cases have been applied. They correspond to ascent and descent from and to the Moon with and without payload combined with full tanks or empty tanks in order to consider the influence of high payload mass or of high acceleration. However, these load cases were in most cases absolutely not sizing. The sizing load case was the launch from Earth. For that purpose load levels considered by NASA [24] have been assumed: in axial direction 5 g and in transversal direction 2 g. Note that in order to uncouple the design of the RLRV from the choice of the AresV launcher chosen in [24], the axial load was increased to 5.5 g to keep the design more conservative. While usually the transversal acceleration is reached at the moment when the product of the angle of attack and the dynamic pressure is maximal, the maximum longitudinal acceleration is usually reached later when a stage of the launcher is getting empty. It was however assumed that both loads occur simultaneously. The design presented hereafter is therefore relatively robust. For the load case in the launcher, it is moreover assumed that only propellant for the descent to the surface of the Moon is available. No payload is considered to be mounted on top of the RLRV, as it would lead to an oversizing of the RLRV structure. Payloads can be either launched on another flight or as a second payload of the launcher with a double launch adapter.

The mass of other subsystems has been estimated based on data available from comparable studies [18]. Two special cases are the thermal insulation of the tank which should be particularly good and landing system for which a margin of 100% was considered. Note that the landing system has been derived from the one of the Apollo Lunar lander where the honeycomb cartridge absorber is replaced by a reusable system such as pressurized metal bellows [25] and [26].

The results of the structural design and of the preliminary sizing of the other subsystems of the lander are plotted in Fig. 6. As expected from experience the structural index (defined as the ratio of the dry mass to the fluid mass) decreases with increasing propellant loadings. It can also be seen that the smaller the tank diameters, the lower the mass. This evolution is well known from rocket stages, as the mass of a bulkhead is relatively high compared to the cylindrical part. However as it can be seen in Fig. 5, some tank designs are getting very long even if they are very light such as the H16d2d2. For larger propellant loading the diameter has to be increased. Even with the largest considered diameters of 4 m for the LH2 tank and 3 m for the LOx tank, the total height of the RLRV is getting above 10 m when the landing legs are deployed.

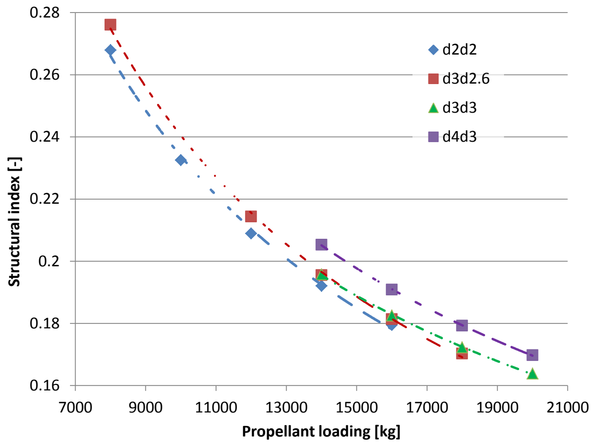

Fig. 6. Evolution of the RLRV structural index (including margins) with the maximum propellant loading

Considering the performances required according to the preliminary selection of payloads presented in [3] and the estimated performances required for the refuelling of the transfer vehicle flying between LEO and LLO/EML1, a maximum propellant loading of 20 tons has been selected. In most cases, the RLRV will not be flying with full tanks, however a larger tank volume increase the flexibility of the RLRV which can perform a large range of mission (see subsection 5.1).

4.4 Selected design:

The selected design of the RLRV is built around a H20 tank with 3 m diameter tanks. The engine has a thrust level of 100 kN. It has a dry mass of just over 3.2 tons including margins. An artist's view can be seen in Fig. 7.

In the figure the RLRV is ready to be launched again. The payload which can be accommodated on the top of the vehicle is not present. Considering the height of the vehicle, 14.5 m, it would be complex to unload the RLRV autonomously or with the help of a system such as NASA's Athlete rover [27]. And it makes not much sense either to carry a system allowing unloading large payload, all the time, on the RLRV. It is sufficient that, for the first mission to a new location, an unloading system such as a crane is transported, folded, in the lower payload bays next to the LOx tank. The crane would be first deployed at a selected location and ballasted with Moon regolith to guarantee its stability. Moon regolith would be gathered by rovers as described in [3]. As seen in the Fig. 7, the RLRV is equipped with wheels. This has several advantages. While it may allow a small lateral velocity at landing, the main use of the wheel would be to allow tugging of the RLRV from its landing position to the location of the crane. The payload would then be unloaded by the crane and loaded on an Athele type rover, which would install it at its final location. The wheels would also allow tugging the RLRV next to the ISPP plant for refuelling and then further away for the next launch. As developed with more details in [3], landing or launching close to existing installations is indeed problematic due to the impingement of dust particles accelerated by the engine exhaust. Note that at the first landing a beacon system such as the one described by Theil et al. [28] will be installed. With such a system Theil demonstrated that landing accuracy better than 10 m (3σ) can be achieved.

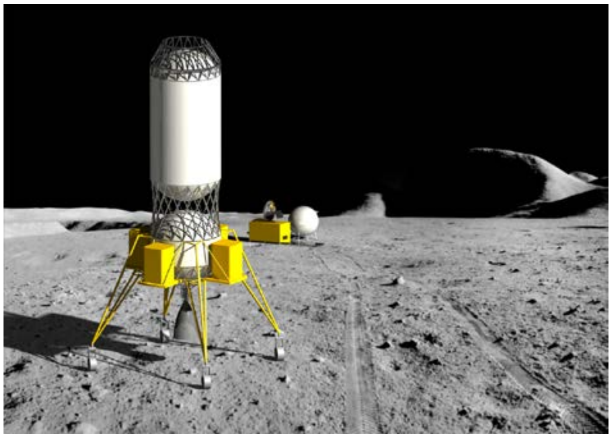

Fig. 7. Artist's view of the selected RLRV design with the ISPP plant in the background

5. Operations

5.1 Missions

While cost saving was the first argument put forward to justify the advantage of a reusable lander and ascent vehicle. Reusability actually offers also flexibility for different missions. During its first mission the RLRV should be launched to LEO by a launch vehicle. Then it would be brought to LLO or EML1 by a reusable transfer vehicle not described in this paper and which design will be performed in more details in a further study. The RLRV would already be loaded with just enough propellant to perform a descent to the Moon with an ISPP plant as payload. Once on the lunar surface the ISPP plant would be deployed automatically and would start producing propellant. Ideally the first landing should be performed at a pole next to a water ice source. In the case of a landing at another point of the lunar surface, only LOx could be produced and enough LH2 should be carried from the Earth and kept at cryogenic state with the help of a cryo-cooler for the next ascent of the RLRV. Once this first step has been achieved, the RLRV is then available for a whole range of missions.

Several missions have been defined and are used to assess the performances of the RLRV.

- Mission 0: (reference mission) the RLRV is landing a mass of 10 tons on the Moon with the minimum mass of propellant (if no LH2 ISPP is available the LH2 for the ascent is carried during the descent)
- Mission 1: the RLRV is landing at the Moon base with as less propellant as possible and no payload (or with the LH2 needed for the next ascent). Once on the lunar surface it will be refuelled with an ISPP plant previously deployed.
- Mission 2a: the RLRV is lifting-off with the maximum propellant loading to carry the heaviest possible payload to LLO or EML1. Enough propellant for a descent (mission 1) is kept.
- Mission 2b: the RLRV is lifting-off with the maximum propellant loading but it is not carrying any payload. Enough propellant for a mission 1 is kept. Left-over propellant is the actual payload and can be transfer to another vehicle.
- Mission 3: the RLRV is performing a descent with full tanks from EML1 or LLO with the heaviest possible payload
- Mission 4: in the case water ice is gathered to produce LOx and LH2 at the poles. The RLRV performs a suborbital flight from the lunar base to the ISPP plant at the pole with the lowest possible propellant loading. The moon base is assumed to be placed at the equator.
- Mission 5a: the RLRV is flying from the pole with full tanks and carry as much propellant as possible in a suborbital flight to the base, assumed to be situated at the equator.
- Mission 5b: the RLRV is taking off with full tanks from the polar ISRU plant towards the base assumed to be situated at the equator. During this mission the maximum payload is carried. The ulterior motive of this mission is to carry propellant as a payload to the base

This list of missions is only a selection of what the RLRV can perform. And except mission 0, they all correspond to extreme missions, allowing deriving the whole range of possible missions.

5.2 Performances

The performances of the RLRV (H20d3d3) for the missions defined in the subsection 5.1, have been calculated with the help of trajectory optimisations and are presented in Table 4 in the case of both LH2 and LOx ISPP. The computed trajectories for two missions can be seen in Fig. 8

Table 4. Performances of the RLRV

Mission	Performance of H20d3d3[kg]		Performance as a mass of
with H2 ISPP	LLO	EML1	
0	7055	10535	Propellant
1	2100	3250	Propellant
2a	21925	11880	Payload
2b	7850	5855	Propellant
3	27565	19050	Payload
4	4525		Propellant
5a	6755		Propellant
5b	10610		Payload

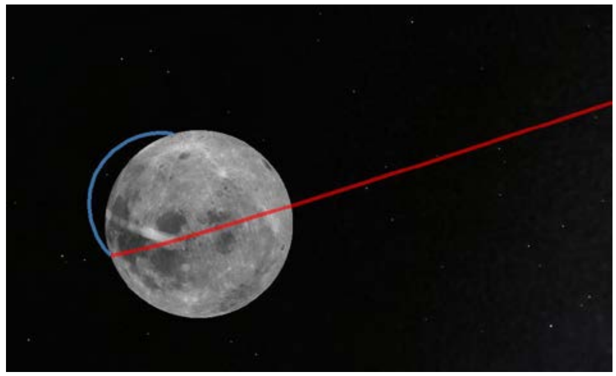

Fig. 8. Computed trajectories for mission 2b to EML1 (in red) and for mission 5a (in blue) (generated with the help of Google Earth)

In the case of the reference mission, it can be seen that only slightly more than 7 tons of propellant are required if the mission is starting in LLO and just more than 10.5 tons are needed if starting from EML1. The corresponding mission starting masses, including a 10 tons payload are 21.8 tons and 25.4 tons for a rendezvous position in LLO and in EML1, respectively. Probably two of such a mission would be needed to establish the ISPP plant according to [3]. This could be performed by two RLRV. Having two RLRV at disposal would indeed increases a lot the capabilities of the infrastructure as they would be able to assist each other. For instance, while one RLRV would perform a mission at its performance limit the second one could carry propellant to refuel it at the end of its mission. Assuming that the RLRV is bringing each time 10 ton payloads to the Moon surface and return to the rendezvous position with just enough propellant to land a new 10 ton payload, the RLRV could bring 50 tons of payload from LLO with an engine lifetime of 3000 s and 100 tons of payload with an engine lifetime of 6300 s. If EML1 is considered the H20 tanks are too small to fit enough propellant for an ascent to EML1 without payload followed by a descent with a 10 ton payload. If the tanks are not extended, only 9.1 tons of propellant can be carried from the Moon. This corresponds to a payload of 7.5 tons for the following descent. Under these conditions, for an engine lifetime of 3000 s, 30 tons can be landed on the Moon. 50 tons are reached after for a lifetime of 5500 s and 100 tons for a lifetime of less than 11000 s.

6. Comparison with Apollo LM and advantages of the RLRV

According to [11], the mass of the Apollo LM including both the descent stage and the ascent stage was around 16.5 tons, of which 4.8 tons correspond to the ascent stage with its propellant. In order to compare the Apollo LM and the RLRV, one can consider that the ascent stage is equivalent to the payload landed by the LM descent stage.

Different payload masses to be landed on the Moon have been selected. The first payload mass is 48 tons and correspond exactly to 10 LM. 100 tons of payload has also been considered as well as 175 tons which is the mass of the habitat defined as the most demanding mission in the ROBEX ALUNIR architecture study [3]. While some modules are heavier than 4.8 tons, it is supposed that the whole habitat can still be transported by the LM and that in the optimistic case that they all have a mass of 4.8 tons. For the RLRV it is not so critic as it can adapt its propellant loading to the mission. Moreover it has been considered that the RLRV can be used 4000 s, after this duration a new RLRV has to be launched from Earth. In order to refuel the RLRV, it was assumed that a 20-ton ISPP plant producing LLOx and LLH2 is installed on the Moon during the first mission of two RLRV. Naturally in the case of the LM no ISPP plant is brought to the Moon. The results of the comparison based on these assumptions are shown in Fig. 9.

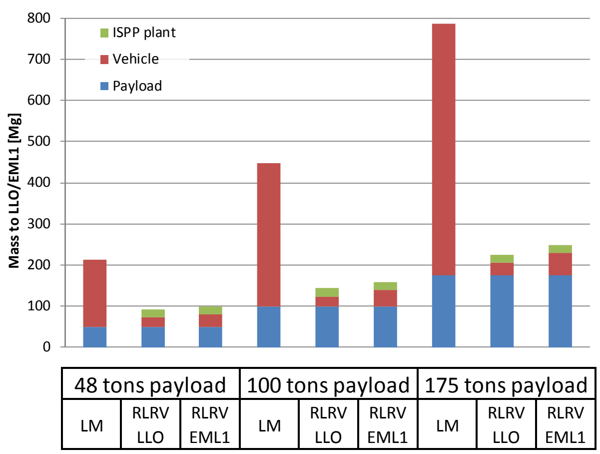

Fig. 9. Comparison of the mass to inject in LLO or EML1 in the case of the LM or the RLRV for given payload masses

While the RLRV flying between the EML1 and the Moon surface transports the payload from a position easier to reach from the Earth than the LM, it appears that in every case the mass to be injected is much lower. For a RLRV operated between the LLO and the lunar surface the difference with the LM is the largest.

In the case of a payload of 48 tons, 213 tons should be injected in LLO if the LM is used. The mass can be reduced by 57% for a RLRV operated in LLO and 53% for a RLRV operated in EML1, despite the ISPP plant.

If the payload mass is increased to 100 tons, the advantage of using the RLRV is increasing. Instead of launching almost 450 tons with the LM, two RLRV operated from LLO would be sufficient and would allow a reduction of the mass injected in LLO by 68%. In the case of the EML1 a third RLRV would be required but still the mass would be decreased by 64%

Finally for the habitat of the ROBEX ALUNIR architecture study, 37 LM would be required in the best case to land 175 tons on the Moon. In addition of having a whole graveyard of used landers next to the base over 785 tons would have to be injected to LLO. This mass is reduced to about 225 tons (-71%) if 3 RLRV are operated from LLO and 250 tons (-68%) if 5 RLRV are operated from EML1.

If the transport from the Earth to the LLO or EML1 is performed as during the Apollo mission, the gain in percentage is exactly the same as for the mass injected in LLO. But if reusability and the use of LLOx and optionally LLH2 are implemented for a transfer stage between LEO and LLO/EML1, the gain will even be strongly magnified, reducing further the mass to be launched from Earth.

7. Conclusions

Classic designs of Moon transportation systems relying on expendable landers propelled by storable propellant pressure-fed engines have shown their capability to perform Moon missions successfully. There are, however, not taking advantage of the numerous technical developments achieved since the 1960s. The long experience accumulated with cryogenic propulsion and newly developed technologies such as the API, the IVF or laser ignition are making the design of a Moon lander propelled by an expander cycle engine running on LOx and LH2 realistic. In the case of the Moon, the LOx/LH2 propellant combination is particularly well adapted to ISPP. Indeed, different studies demonstrated that LOx and under certain conditions LH2 can be produced on the Lunar surface either from regolith or water ice. The use of LOx/LH2 as propellant in combination with ISPP enables the introduction of reusability. A re-usable Moon transportation system propelled by an expander cycle engine running on LOx and LH2 can be further optimized through the selection of the position of the rendezvous between the lander and the transfer stage. While LLO is still considered in most designs today, EML1 offers many advantages such as the accessibility of the whole Moon surface. The origin of the propellant and especially of the LH2 has also an influence on the system. For most locations on the Moon, it cannot be produced in-situ and it should be brought either from the Earth or from the poles of the Moon. The transport from the pole to another location on the Moon surface can then be achieved by a suborbital flight.

Based on these findings a re-usable vehicle called RLRV, propelled by an expander cycle engine running on LOx/LH2 and relying on ISPP has been the object of a preliminary design. In its reference mission, this vehicle should land a payload of 10 tons on the lunar surface. Due to its reusability it is however able to perform a large range of missions such as landing small and large payloads, launching payload to LLO or EML1, but also performing sub-orbital flights. The thrust level and the structural architecture of the RLRV have been selected in order to optimise the performances and capabilities of the RLRV. The selected engine has a thrust of 100 kN. A separated bulkhead design with one tank for each of the propellant has been selected. The feed system and the RCS have been designed in such way that they are not limiting the life expectancy of the RLRV thanks to consumables which can be refuelled on the Moon. While the performance assessment of the RLRV showed that the vehicle is very flexible and can adapt to very different missions, a comparison with the Apollo LM for various payload masses indicated that a very important gain can be reached. For instance in the case of the habitat predesigned in the ROBEX project, the use of the RLRV could cut by 71% the mass to be injected in LLO from the Earth.

Future works should concentrate on refining further the mission scenarios for the RLRV and study under which conditions the use of propellant produced on the Moon could be advantageous to propel a re-usable transfer stage.

Acknowledgements

This work has been performed within the ROBEX project (www.robex-allianz.de/en/), coordinated by the Alfred Wegner Institute and funded by the Helmholtz Association of German Research Centers.

The author would also like to acknowledge the contribution of Mr. Gero Förster for the CAD model of the RLRV.

References

[1] D. Wilde, P. Kyr, T. Diedrich, C. Bank, K. Pietsch, From ISS to Moon and beyond: European Transportation Systems for Space Exploration, AIAA 2009-6741, Space 2009, Pasadena, USA, 2009, 14-17 September.

[2] B. M. Birkenstaedt, J. Hopkins, B.F. Kutter, F. Zegler, T. Mosher, Lunar Lander Configurations Incorporating Accessibility, Mobility, and Centaur Cryogenic propulsion Experience, AIAA 2006-7284, Space 2006, San Jose, USA, 2006, 19-21 September.

[3] E. Dumont, Design of a Modular Transportation System for Future Lunar Robotic Missions, 2015-k-24, 30th International Symposium on Space Technology and Science, Kobe-Hyogo, Japan, 2015, 4-10 July.

[4] J.B. Hopkins, Comparison of Propulsion Options for a Lunar Lander Ascent Stage, AIAA 2005-6740, Space 2005, Long Beach, USA, 2005, 30 August – 1 September.

[5] R.M. Zubrin, The Design of Lunar and Mars Transportation Systems Utilizing Extraterrestrial Resources, IAA 92-0161, 43rd congress of the International Astronautical Federation, Washington DC, USA, 1992, 28 August – 5 September.

[6] A. Braukhane, M. Kreutz, E. Ksenik, C. Lange, L. Richter, O. Romberg, Velox – A demonstration facility for Lunar Oxygen Extraction in a Laboratory Environment, GLUC-2010.2.8.A.6, Global Lunar Conference 2010, Beijing, China, 2010, 31 May-3 June.

[7] D.R. Williams, Ice on the Moon: A Summary of Clementine and Lunar Prospector Results, 10 December 2012, http://nssdc.gsfc.nasa.gov/planetary/ice/ice_moon.html (accessed 23.08.2016)

[8] N.N., Air Liquide Gas Encyclopedia, http://encyclopedia.airliquide.com/encyclopedia.asp (accessed 23.08.2016).

[9] N.N. Grumman, Apollo News Reference, Lunar Module: Quick Reference Data

[10] J. Barr, ACES Stage Concept: Higher Performance, New Capabilities, at a Lower Recurring Cost.

[11] J. Connolly, Kickin'up some dust, presentation, NASA/JSC, USA, 2007, 22 February.

[12] J.F. LeBar, E.C. Cady, The Advanced Cryogenic Evolved Stage (ACES) A Low-Cost, Low-Risk Approach to Space Exploration Launch, AIAA 2006-7454, Space 2006, San Jose, USA, 2006, 19-21 September.

[13] N.N. ULA, Transportation Enabling a Robust Cislunar Space Economy, 2016, 8 June.

[14] S. Stappert, E. Dumont, Reusability of launch vehicles by the method of SpaceX, SART TN-007/2016, 2016.

[15] E.L. Axdahl, J.A. Gabler, M.S. Grimes, R.A. Lugo, P.R. Chai, M.R. Long, R.A. Rowland, Reusable Lunar Transportation Architecture Utilizing Orbital propellant Depots, AIAA 2009-6711, Space 2009, Pasadena, USA, 2009, 14-17 September.

[16] M.R. Bobskill, M. Lupisella, D. Reeves, J. Williams-Byrd, M. Simon, K. Larman, H. Thronson, R. Lewis, Earth-Moon L1 / L2 Infrastructure – What Role Does It Play?, Human Exploration Community Workshop on the Global Exploration Roadmap, 2011, 15 November.
[17] D. Jones, Cryogenic Propulsion Stage, NTRS, 2011.
[18] JSC Advanced Design Team, L1 Lunar Lander Element Conceptual Design Report, EX15-01-092, 2000, November.
[19] N.N. NASA, Lunar Capability Concept Review (LCCR) Transportation Systems Only, 2008, June 18-20.
[20] J. Deeken, D. Suslov, N. Rackemann, A. Preuss, Combustion Performance and Stability of a Porous Injector Compared with a State-of the Art Coaxial Injector, SP2014-2969154, Space Propulsion 2014, Cologne, Germany, 2014, 19-22 May.
[21] M. Börner, C. Manfletti, M. Oschwald, Laser Re-ignition of a Cryogenic Multi-Injector Rocket Engine, 6th European Conference for Aeronautics and Space Sciences, Krakow, Poland, 2015, 29 June – 3 July.
[22] F. Zegler, Development of an Integrated Propulsion and Power System for Long Duration Cryogenic Spaceflight, AIAA 2012-5302, Space 2012, Pasadena, USA, 2012, 11-13 September.
[23] C. Manfletti, Laser Ignition of an Experimental Cryogenic Reaction and Control Thruster: Ignition Energies, Journal of Propulsion and Power Vol. 30, No.4 (July-August 2014), 952-961.
[24] K.C. Wu, J. Antol, J.J. Watson, R. J. Saucillo, D.D. North, D.D. Mazanek, Lunar Lander Structural Design Studies at NASA Langley, AIAA 2007-6137, Space 2007, Long Beach, USA, 2007, 18-20 September.
[25] W.F. Rogers, Apollo Experience Report – Lunar Module Landing Gear Subsystem, NASA TN D-6850, June 1972.
[26] J. Trautwein, Pressurized Metal Bellows Shock Absorber for Space Applications, University of Central Florida, Master Thesis, 2015.
[27] C. Culbert, Lunar Surface Systems Project Overview, U.S. Chamber of Commerce Programmatic Workshop on NASA Lunar Surface Systems Concepts, Washington DC, USA, 2009, February.
[28] S. Theil, L. Bora, Beacons for supporting lunar landing navigations, CEAS Space Journal (2016), 1-19.