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Architectural Analysis for the Gateway

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Abstract

As humans move out of the low earth orbit and into cis-lunar-space new challenges must be faced. This paper analyses and discusses a possible architecture, from the orbit selection to the life support system and transport infrastructure of the cis-lunar space station Gateway. Initially, the existing knowledge and previously presented concept from NASA will be disregarded on purpose. The final part of the analysis is the identification of differences between the newly developed architecture and the currently proposed architecture from NASA. Based on the differences advantages as well as disadvantages of both concepts are then discussed.

The first part of the architecture is the life support system, which is analysed with a combination of multi-criteria analysis and Equivalent System Mass to select an optimal life support system. Additionally, the required initial launch mass of the systems as well as the required continuous resupply mass is calculated based on the selected life support system. The next step is the selection of a suitable orbit, which is performed by using the System Tool Kit to develop a database for the overall delta v demand of the mission for different orbits. The optimization considers the calculated mass requirements from the initial analysis as transportation requirements for the Earth – Gateway orbit. In addition, further mission to the lunar surface can be defined to identify trade off points between different orbits that might be preferable for different combinations of Earth – Gateway, Gateway – Moon flights. For the further analysis a suitable orbit which minimizes the overall Δv demand of the architecture is selected. Based on the calculated required Δv and transport requirements possible launch vehicles and their suitability to the mission are discussed. In addition, an interface minimization approach which considers the number and complexity of interfaces and minimizes the overall complexity of interfaces between different international ground control centers and the individual console positions of the ground control centers is proposed to develop a concept of operations. The developed architecture is then compared to the currently proposed architecture of Gateway where possible and applicable.

Keywords: Gateway, ECLSS, Orbit Dynamics, Equivalent System Mass, Multi Criteria Analysis

Nomenclature

m	Mass
Δv	Required velocity change for a specific orbital maneuver
v_e	Exhaust velocity of the engine

Acronyms/Abbreviations

4BMS	4 Bed Molecular Sieve
ACS	Atmosphere Control System
AES	Air Evaporation System
AR	Air Revitalization
CAMRAS	CO_2 And Moisture Removal Swing-Bed
DOI	Descent Orbit Injection
DSM	Design Structure Matrix
ECLSS	Environmental Control and Life Support System
EDC	Electrochemical Depolarization Concentration
ESM	Equivalent System Mass
ISP	Specific Impulse
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LiSTOT	Life Support Trade Off Tool
MCA	Multi Criteria Analysis
MF	Multi Filtration
mt	metric tons
ORU	Orbital Replacement Unit
SAWD	Solid Amine Water Desorption
SFWE	Static Feed Water Electrolysis
SPWE	Solid Polymer Water Electrolysis
THC	Temperature & Humidity Control
TIMES	Thermoelectric Integrated Membrane Evaporation System
TLI	Trans Lunar Injection
TRL	Technology Readiness Level
VCD	Vapor Compression Distillation
VPCAR	Vapor Phase Catalytic Ammonia Removal
WRM	Water Recovery Management

1. Introduction

The goal of this paper is to perform an independent analysis of the Gateway architecture including the life support system, orbit selection and transport concept. For this reason, the angelic halo orbit that was selected for Gateway recently [1] as well as some additional available information regarding environmental control and life support systems (ECLSS) will be disregarded initially. Instead only the estimated volumes and intended number of crew members will be used in the analysis of the Gateway [2, 3].

Based on these assumptions an ECLSS design is developed in chapter 3.1 which minimizes the required resupply while adhering to other constraints like schedule and reliability. Chapter 3.2 provides an overview of the available launch vehicles and cargo spacecrafts. In chapter 3.3 different possible orbits and their Δv requirements are analysed. Together with the required resupply calculated in chapter 3.1 an optimal orbit for different assumptions regarding the number of lunar surface missions is derived in chapter 3.4. Chapter 3.5 then discusses the mission architecture regarding possible launch vehicles and cargo spacecrafts. Finally, chapter 3.6 will discuss an approach to optimize the interfaces between different mission control centers based on a design structure matrix approach.

2. Material and methods

For the ECLSS analysis the Life Support Trade Off Tool (LiSTOT) as introduced in [4–6] is used. It contains a database of ECLSS hardware and combines a multi criteria analysis (MCA) and an equivalent system mass (ESM) approach to compare different ECLSS hardware for a mission scenario and select an optimal combination for it. Optimal depends on the specific user criteria which will be discussed in chapter 3.1.

For orbit analysis the System Tool Kit [7] and MATLAB [8] are used to simulate different possible orbits for the Gateway and calculate Δv requirements for different manoeuvres.

The operations concept is analysed with a design structure matrix approach according to [9].

3. Theory and calculation

3.1. ECLSS Design

In order to design an optimal ECLSS in LiSTOT it is necessary to define the mission scenario and the corresponding values for the trade-off analysis.

Table 1. Mission Scenario Assumptions for the Gateway ECLSS Trade Off Analysis

	Value	Unit
Crew Size	4	-
Campaign Duration	900	Days
Number of Modules	7	-
Total pressurized Volume	125	m³
Exercise per Day	0.5	h

Note that the campaign duration of 900 days is the cumulative sum of crewed days over the whole campaign with one 60-day long mission per year. As maintenance strategy for the trade-off three ORU per technology are assumed to be sufficient for save operations over the entire campaign. Because reliability data for many systems is not available, it is not possible to calculate the required number of spares based on a “mean time between failure” approach. Table 2 and Table 3 provide an overview of the assumed parameters for the analysis.

Table 2. Assumed trade-off parameters for the analysis with ESM values from [10]

		Value	Unit
ESM Equivalency Factors	Volume	35.9	kg/m³
	Power	60	kg/kW
	Cooling	55.4	kg/kW
	Crew Time	0.8	kg/h
MCA Parameters	Minimum TRL	5	-
	Weights	all 1	-

Table 3. Crew metabolic loads per crew member based on values from [11]

	Value	Unit
Oxygen	0.82	kg/d
Carbon Dioxide	1.04	kg/d
Potable Water	2.52	kg/d
Hygiene Water	6.8	kg/d
Urine	1.6	kg/d
Sweat	1.92	kg/d
Feces	0.12	kg/d
Food	1.51	kg/d
Heat	12	MJ/d

Using these values trade-offs between different subsystems and different overall ECLSS compositions (from an open loop to a closed loop bio-regenerative system) were performed to select the optimal ECLSS architecture. The detailed trade-offs of the different architectures and systems are discussed in detail in [6]. Here only a discussion of the most interesting results will be provided.

Regarding the overall architecture a partially closed loop approach proved to be the best solution as shown in Fig. 1.

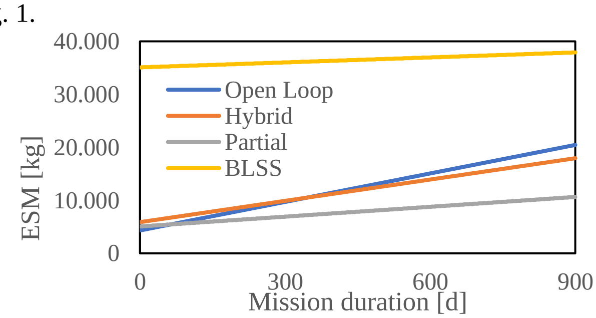


Fig. 1. Comparison of ESM for different ECLSS Architectures

It is noteworthy that the breakeven point between the partially closed system and the hybrid system is after 2845 days while the breakeven between the partially closed and bioregenerative ECLSS is after 9821 days. This suggests that for mission durations on a similar time scale as the ISS operations the hybrid system would be better suited. However, for the considered 900 days of crewed operation the partially closed physical chemical life support system was by far the best overall option. Therefore, some of the interesting trade-offs regarding its components are discussed here. For example, for CO_2 reprocessing the Bosch reactor was selected as it resulted in a lower resupply mass and in an overall lower ESM after 284 days in the trade-off because of the higher O_2 recovery rate compared to the Sabatier.

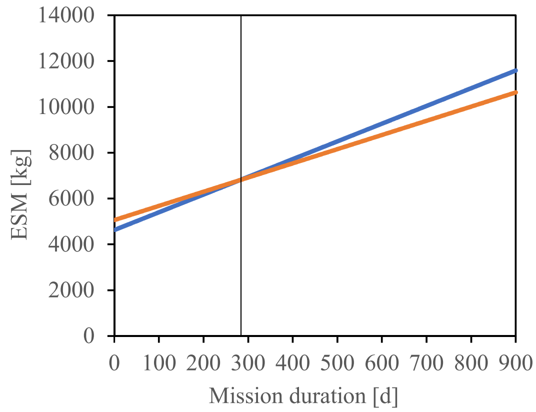


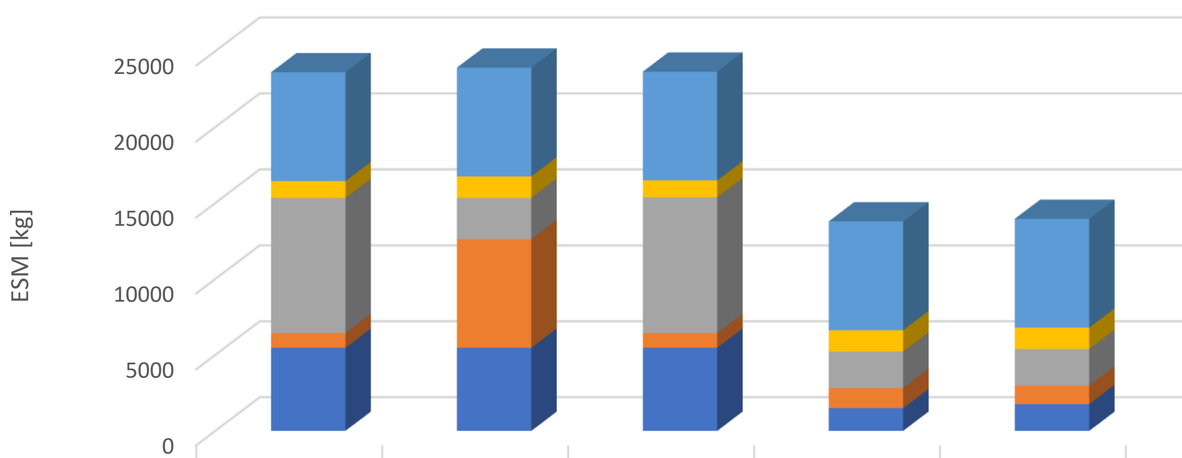
Fig. 2. Comparison of the ESM for a Bosch Reactor (Orange) and a Sabatier (Blue)

The calculation was performed using the following values for the Bosch and Sabatier system.

Table 4. Performance Data for the Bosch and Sabatier System based on values from [12, 13]

	Bosch	Sabatier	Unit
Mass	268.7	125	kg
Volume	0.31	0.15	m³
Power	1182.8	46.5	W
Cooling	324.8	240	W
Crew Time	24	0	h
Reliability	0.99	0.99	-
TRL	6	9	-

After further trade-offs between different subsystem various combinations of CO_2 removal and reprocessing systems and water recovery systems were compared to arrive at an optimal system level architecture for the Gateway ECLSS. The ESM results of these trade-offs are shown in Fig. 2 which also shows how the different systems contribute to the overall ESM. The open loop option to use LiOH for CO_2 removal is also shown in the figure. For CAMRAS as CO_2 removal option currently no CO_2 reprocessing is considered but CAMRAS by itself also has a higher ESM than a solid amine water desorption (SAWD) system. Overall, the best combination of technologies according to both the ESM and multicriteria analysis was the SAWD with a Bosch reactor and a Thermoelectric Integrated Membrane Evaporation System (TIMES) for water recovery.



	CAMRAS + AES	LiOH + AES	CAMRAS + TIMES	SAWD + TIMES + Bosch	EDC + TIMES + Bosch
Food/Clothes	7162	7162	7162	7162	7162
THC	1115	1410	1115	1410	1410
WRM	8897	2726	8939	2405	2405
AR	965	7148	965	1324	1216
ACS	5468	5468	5469	1491	1766

Fig. 3. ESM Values for different combinations of subsystems

Using preliminary dynamic calculations of the O_2, CO_2 and humidity levels in the Gateway a second design cycle for the selected ECLSS was performed to further optimize it. The final design requires the following resupply masses.

Table 5. Required Resupply masses for the designed ECLSS for 4 crew members

	Value	Unit
O_2 Generation or Supply	29.2	g/d
Nitrogen Supply	156	g/d
Water Supply	0	g/d
Food	5640	g/d
Clothes	421	g/d
CO_2 Removal	19.4	g/d
CO_2 Reduction	29.1	g/d
Wastewater Filtration	1414.3	g/d
Urine Processing	16	g/d
Electrolysis	0	g/d
Total	7725	g/d

Surprising is probably that no water resupply is required. That is the case because the water content of the provided food (~0.7 kg per crew member and day [11]) and the generated metabolic water (~0.4 kg per crew member and day [11]) are higher than the losses of water from oxygen generation and waste water treatment. To produce the required oxygen mentioned in Table 3 only ~920 g of water is required for electrolysis. A detailed overview of the different mass flows in the ECLSS is provided in Fig. 4.

3.2. Launch Vehicle and Spacecraft Performance

This chapter provides basic information regarding the currently available launch vehicles and cargo spacecrafts which are feasible for cis-lunar resupply missions. For the cargo spacecrafts, the relevant information required to calculate possible payload masses are the spacecraft dry mass, the available fuel and the specific impulse (ISP). The ISP is defined as the exhaust velocity v_e of the engines divided with the standard acceleration of earth gravity and provides a measure for the fuel efficiency of the engine.

Table 6. Assumed cargo spacecraft values

Spacecraft	Dry Mass [kg]	Fuel Mass [kg]	ISP [s]
Cygnus	1923[14]	800[14]	300
Dragon	4200[15]	1290[15]	234 [15]
HTV	9068 [16]	2432 [16]	300
Orion	14197[17]	8600 [18]	315 [19]
Progress M	4050 [20]	900 [20]	305 [20]

For Cygnus and HTV no information regarding the ISP could be found but an estimate based on the used propellant and Ref. [21] as well as other engines using the same propellant types was made. For Dragon no information for the Draco thruster used by the uncrewed variant could be found and instead a value for the Super-Draco thruster of the crew variant is used. Based on these values the available Δv of these spacecrafts can be calculated using the Ziolkowski Equation.

$$\Delta v = v_e \cdot \ln\left(\frac{m_{initial}}{m_{final}}\right) \tag{1}$$

Or based on this equation the possible payload mass for a required Δv can be calculated using the following equation.

$$m_{payload} = \frac{m_{dry} + m_{fuel} - e^{\frac{\Delta v}{v_e}} \cdot m_{dry}}{e^{\frac{\Delta v}{v_e}} - 1} \tag{2}$$

For launchers the following values are used in the analysis.

Table 7. Assumed Launch Vehicle Values from [22–24]

	TLI Payload [kg]
Falcon 9 (Drone Ship Recovery)	3380
Atlas V (551)	6175
Ariane 64	8500
Falcon Heavy (recovery)	10300
Delta IV (Heavy)	10300
Falcon Heavy (expendable)	15190
SLS Block 1	26000
SLS Block 1B	37000
SLS Block 2	45000

It is noteworthy that the Trans Lunar Injection (TLI) payload of the falcon rocket family is lower than the payload values to Mars SpaceX provides on their homepage. However, as the values in the table for these rockets are calculated using the NASA vehicle performance estimator using a C3 energy of -0.6 km^2/s^2 to calculate the TLI payload [22], these values are used for the analysis as Gateway missions would be NASA missions. Note that the chosen C3 value for the TLI enables low energy transfer and a direct transfer could also be achieved with -2 km^2/s^2 [25] but the difference in payload is fairly small (275 kg for a Delta IV Heavy) and for Falcon Heavy no data was available for -2 km^2/s^2.

Other available launch vehicles, like the H-IIB from Japan, have a similar or smaller capability to the Falcon 9 in Table 7. Comparing the Payload capacity of the launchers and the available spacecrafts as described in Table 6 shows that these smaller rockets could at most transport the Cygnus spacecraft with a maximum payload of ~600 kg. Therefore, other launch vehicles are not considered. The option to launch the cargo spacecraft into LEO and then performing the TLI from LEO using a separate engine stack was not considered here as currently no platform for such a transfer is available.

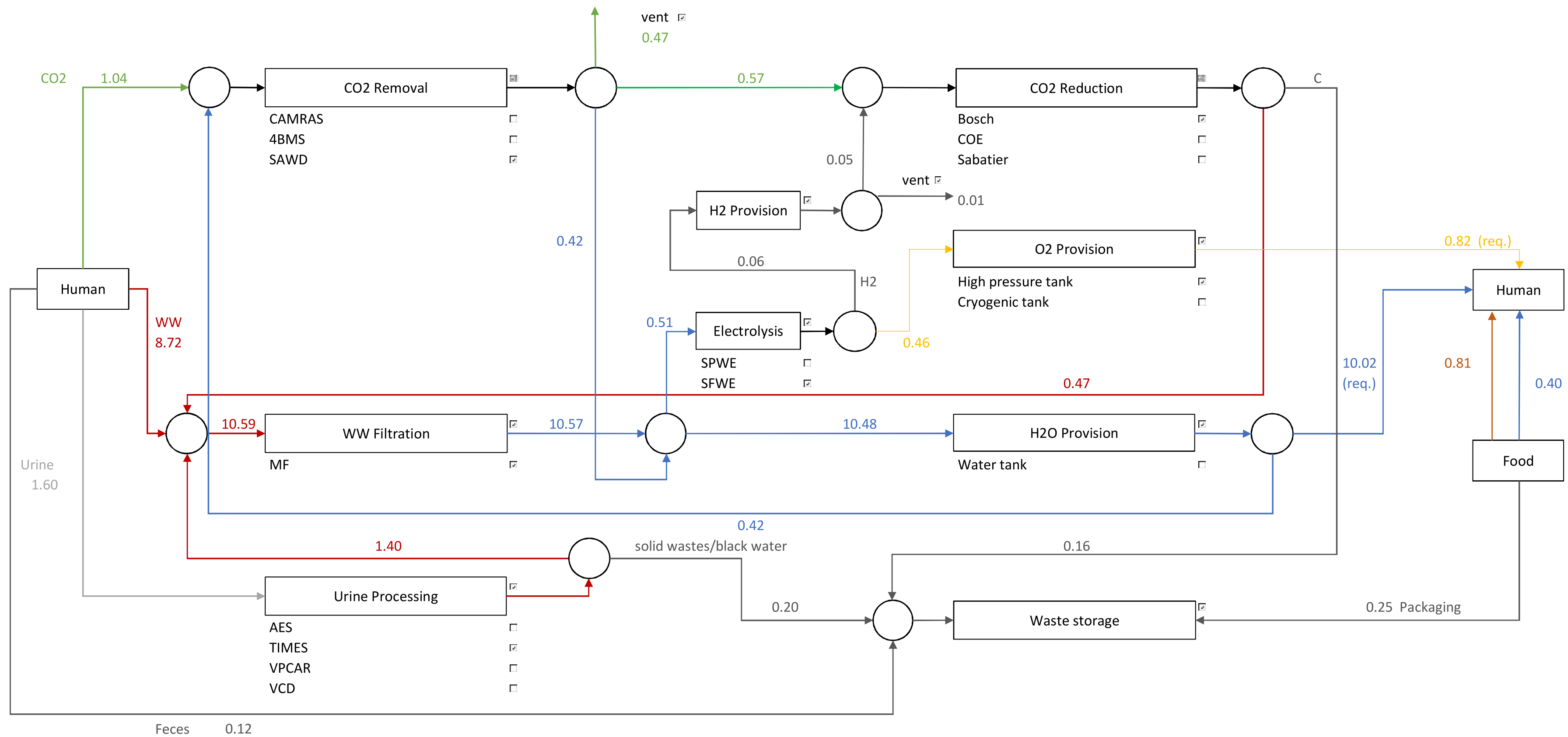


Fig. 4. Partially closed loop ECLSS Schematic from LiSTOT with the different available technology options. The shown numbers are mass flows in kg/day for the final ECLSS design consisting of SAWD, TIMES, Bosch, SFWE, TIMES and basic storage tanks for all remaining parts.

3.3. Orbit Dynamics

To create an optimal architecture for the Gateway it is necessary to have accurate information on the different orbits and the required Δv for different orbital maneuvers. Near rectilinear halo orbits, a subset of the halo orbit families, have been a strong candidate for the Gateway. Therefore, this paper focuses on the halo orbit families originating from the libration points L1 and L2. Past research covered the calculation and analysis of halo orbits. Most prominently by Farquhar and Kamel [26], Breakwell and Brown [27] and Howell [28], who employed analytical and numerical methods to compute halo orbits. For this paper, we find single orbits through a multiple shooting framework. A good initial guess is mandatory for a converging solution due to the high nonlinearity in the equations of motion. Starting with the linearized equations of motion of the circular restricted three-body problem, solutions indicate the existence of periodic planar Lyapunov orbits but not the existence of periodic three-dimensional halo orbits. For larger amplitudes of Lyapunov orbits, nonlinear effects dominate and eventually allow for halo orbits to bifurcate from Lyapunov orbits at a specific amplitude [29]. This principle is applied numerically to find halo orbits.

A linear Lyapunov orbit in close vicinity of the libration point serves as initial guess to find a corresponding Lyapunov orbit in the nonlinear model with multiple shooting. The shooting scheme constrains the amplitude and solves for components of the initial state to yield a closed orbit. From there on, the amplitude constraint is iteratively increased to compute the entire Lyapunov orbit family. To detect the amplitude at which the bifurcation occurs, the stability of each Lyapunov orbit is determined by assessing the eigenvalues of the state transition matrix evaluated after one orbital period. This procedure follows the stability analysis of halo orbits performed in [28]. Once the eigenvalues indicate the bifurcation, the initial position of the critical Lyapunov orbit is deviated out of the horizontal plane by a small distance and vertically constrained. Following the same procedure as for computing Lyapunov orbits, the initial state of the orbit is solved with multiple shooting to obtain a periodic solution. Subsequent halo orbits are found by geometrically constraining the distance between the position of the lastly calculated orbit and the subsequent orbit. As soon as the point of the orbit closest to the Moon reaches its surface, the iteration is aborted. The result of the procedure is a look-up table of initial states for a large number of halo orbits. Fig. 5 shows the generated halo orbits originating from the two Lagrangian points. For these orbits the required Δv for different maneuvers and their transit time was calculated. Parker and Born [30] provide a survey of direct transfers between Earth and halo orbit on a broad parameter space.

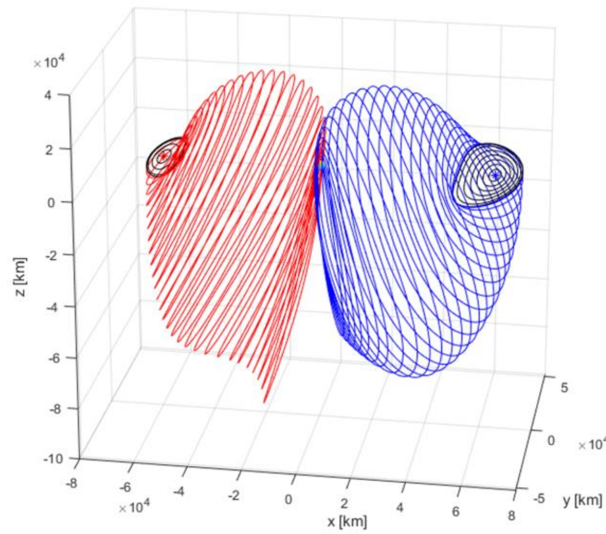


Fig. 5. Visualization of the analysed Halo Orbits. Red shows possible L1, blue possible L2 Halo orbits

For the investigation in the present paper, the range of strategies for transfers between Earth and halo orbit is narrowed down to the following methodology. Perilune and apolune states of the previously computed halo orbits are transferred to STK. By backward propagation, a target sequence solves for the magnitudes of two tangential maneuvers performed on a LEO at an altitude of 185 km and either at the perilune (L1) or apolune (L2) of the halo orbit. For transfers from halo orbits to a polar LLO, two strategies with two variants each are employed. The following abbreviations are used in the following figures:

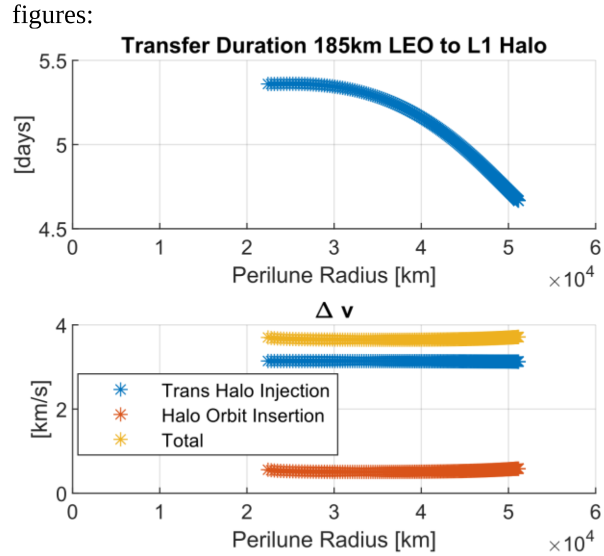


Fig. 6. Δv and duration for Earth to L1 Halo transfer

T_1: A non-tangent Descent Orbit Injection (DOI) directly into transfer orbit to a polar LLO.

T_2: A tangent DOI to a transfer orbit to LLO, the inclination change to achieve a polar LLO is then performed in the LLO by a separate maneuver.

The letters “a” and “p” represent the location on the orbit where the maneuver is performed. “a” represents the apolune, while “p” represents the perilune.

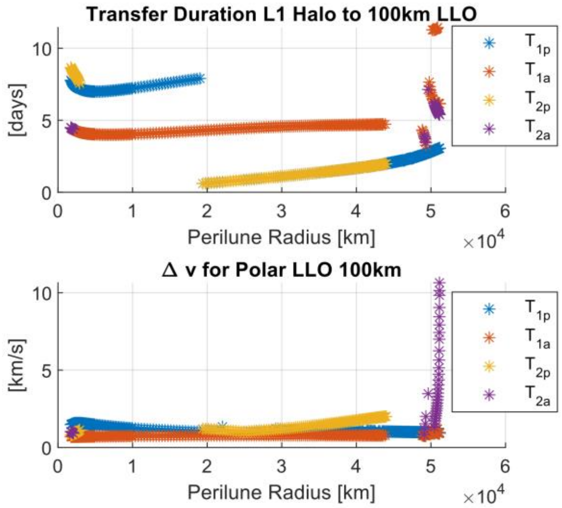


Fig. 7. Δv and duration for L1 Halo to LLO transfer

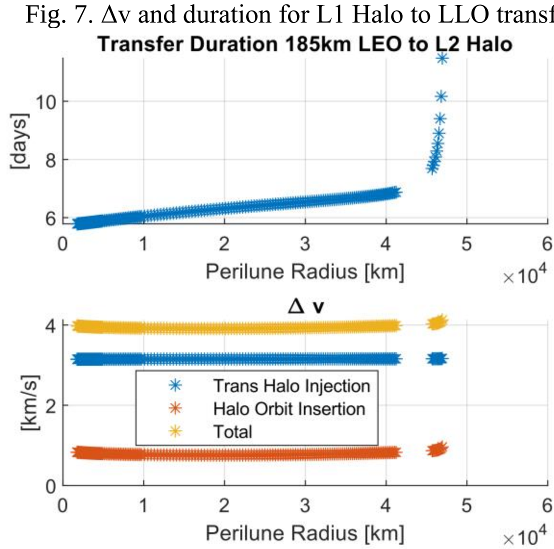


Fig. 8. Δv and duration for Earth to L2 Halo transfer

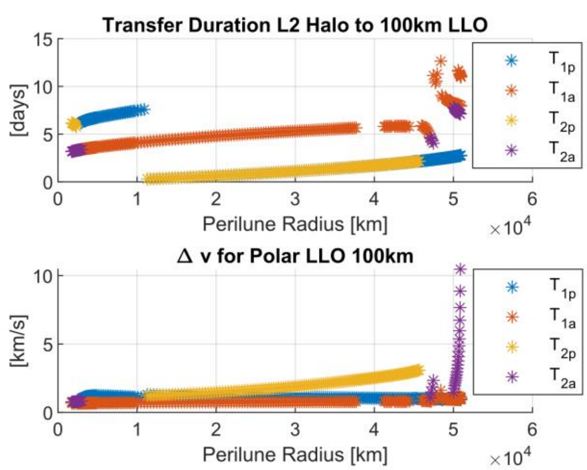


Fig. 9. Δv and duration for L2 Halo to LLO transfer

The missing data points from these plots are cases where the initial guess was not accurate enough to find a converging solution. However, as can be seen in the plots the orbits with the smallest Δv are part of the solved orbits and the optimal solution is likely close to the minimal Δv. To fill these gaps in the analysis is currently future work as it was not possible to achieve it in time for the release of this paper. In addition, the current transfers do not consider lunar gravity assists as possibility to decrease the Δv. It is planned to increase the database of possible orbit transfers to include all of these options and include other possible orbits, like the distant retrograde orbit and then reiterate the analysis to arrive at a final conclusion. For this paper, the preliminary results are used to find an optimal solution within the provided parameter space.

More detailed information on the orbital calculations can be found in Ref. [31].

3.4. Optimal Orbit Selection

Using the values from the previous chapter the optimal orbit for Gateway can be selected based on the required masses that must be transported to the different locations. For the analysis the optimal case is the one where the total mass that must be transported (consisting of payload mass, fuel mass, ECLSS mass, crew mass and Gateway mass) is minimal.

The ECLSS is assumed to scale with 5 kg per day and crew member for the transit durations, assuming that Orion uses CAMRAS as regenerative CO_2 removal system and supplying O_2, water and food from storage based on the values from Table 3.

For the orbit selection Cygnus is used to estimate the required spacecraft masses and fuel masses to transport hardware into lunar orbit. Therefore, all hardware is assumed to be transported by Cygnus, even the modules and the lunar landers. Cygnus was used for this stage of the analysis as it is vehicle with the smallest dry mass and therefore incrementing the number of Cygnus vehicles results in less pronounced steps in the created mass data. Since the Δv of the different vehicles is nearly identical the amount of required fuel for the transport is also nearly identical therefore this approach is a valid estimation of the total campaign mass. A detailed consideration of possible combinations for cargo spacecrafts and launch vehicles is then performed only for the selected orbit in chapter 3.5. To calculate the required number of total flights, first the possible payload for Orion is calculated and only the remaining payload is transported with Cygnus. For each lunar landing a payload to LLO of 25 mt is assumed, which is considered to include fuel and engines for the surface landing and subsequent ascent back to the halo orbit. These 25 mt must be transferred into a polar LLO for which gear ratios were also estimated based on Cygnus. The Gateway mass is assumed to be 60 mt with in total 15 crewed missions with 4 crew members and a stay of 60 days per mission (which adds up to 900 days of crewed operation over 15 years). The ECLSS mass which must be prepositioned was calculated to 10.7 mt from LiSTOT. Resupply masses are according to Table 5. In addition, 2 mt of payload per mission to the Gateway itself are assumed.

Since the orbit analysis does not yet cover all possible options, especially the return option for crewed missions, the additional assumption of 30 m/s Δv for orbital rendezvous manoeuvres and course corrections as implemented as well as a return Δv from all Halo orbits of 700 m/s based on [32].

The results of this analysis are summarized in Fig. 10 and Fig. 11 which show different optimal L1 Halo orbits for different numbers of lunar surface expeditions.

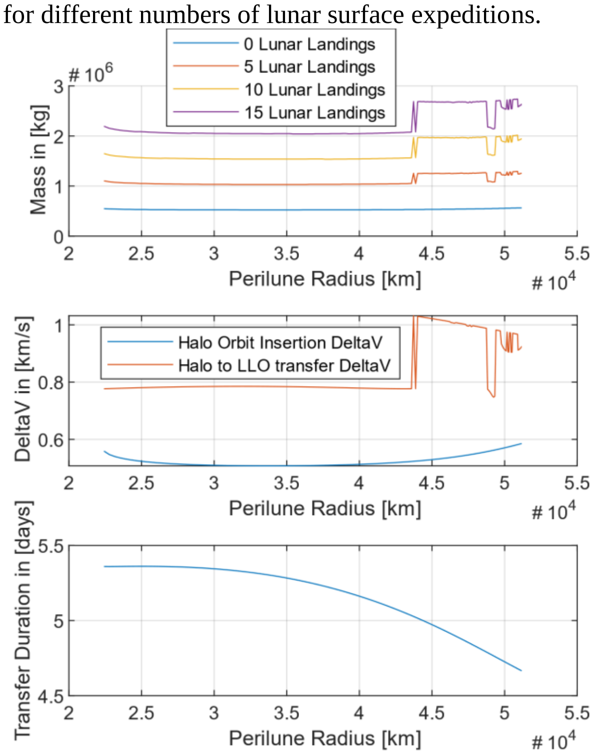


Fig. 10. Total Campaign Mass, minimal Δv and transfer durations for L1 Halo orbits

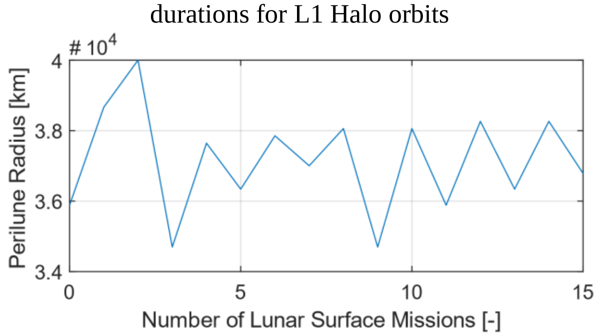


Fig. 11. Optimal L1 Halo Orbits

For the L2 Halo family, unfortunately no possible solution could be found as the direct transfers to L2 require more Δv are not within the possible envelope for Orion. The Δv and transfer durations for L2 Halos are shown in Fig. 12.

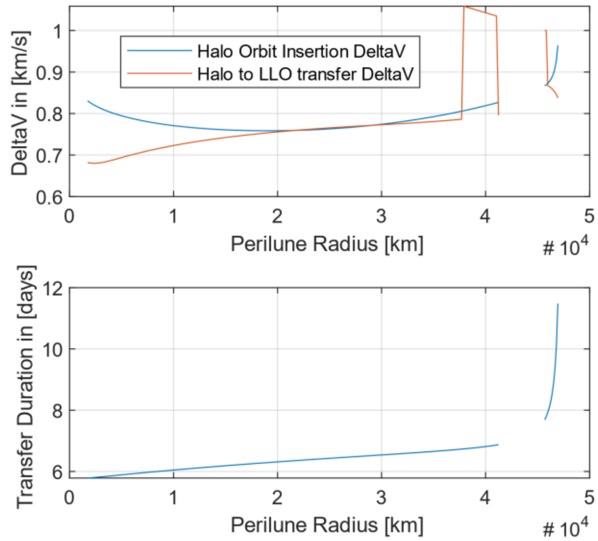


Fig. 12. L2 Halo orbit minimal Δv and transfer durations

3.5. Cargo Spacecraft and Launch Vehicle Selection

For this discussion the results for 6 lunar surface missions from the previous chapter will be used. The Δv required for this orbit for a one- way cargo mission including the 30 m/s additional Δv are 540 m/s. The following table shows the payload capacities for the different cargo spacecrafts with their base fuel and with 25% increased fuel capacity.

Table 8. Cargo Spacecraft Payload Capacities

Spacecraft	Payload normal [kg]	Payload +25% fuel [kg]
Cygnus	2052	3046
Dragon	678	1898
HTV	3016	6037
Progress M	504	1642

Orion is not part of Table 8 because it is assumed that Orion is only used for crewed missions which require a return to Earth. For this case Orion has a payload capacity of 3227 kg. The total masses that must be transported are summarized in Table 9.

Table 9. Transported Masses to Gateway Orbit

	Value	Unit
Lunar Landing Equipment	125	mt
Transport Landers into LLO	134	mt
Crew Mass	4.9	mt
Transit Resupply	4.1	mt
Gateway Resupply	7	mt
ECLSS Mass	11	mt
Gateway Module Mass	60	mt
Gateway Payload Mass	30	mt
Total	376	mt

Cargo spacecraft can only carry a certain part of the considered masses. E.g. the modules, lunar landers and engines to transport the lunar landers cannot be flown with cargo spacecrafts. Therefore, 57 mt of cargo remain which must be transported by cargo spacecrafts. Orion can carry 48 mt of payload with the crewed flights, which leaves only 9 mt of required resupply by cargo spacecrafts. Using the values from Table 8 the following required flights can be calculated.

Table 10. Required Cargo Spacecraft Flights

Spacecraft	Flights normal [-]	Flights +25% fuel [-]
Cygnus	4.4	3
Dragon	13.2	4.7
HTV	2.9	1.5
Progress M	17.9	5.5

The values are not rounded to integer values to show how well used the last flight would be. The following values for the total spacecraft mass including fuel and payload can be calculated.

Table 11. Total Cargo Spacecraft Mass

Spacecraft	Total Mass normal [kg]	Total Mass +25% fuel [kg]
Cygnus	4775	4975
Dragon	6168	6491
HTV	14516	15124
Progress M	5454	5679

In general, increasing the fuel capacity by 25% seems feasibly for all cargo spacecrafts, as the payload capacity to the moon including the additionally fuel would still be smaller than the current LEO payload capacity, which means that likely the required additional space is available on the spacecrafts without major redesigns. However, in principle the spacecrafts are also capable of supplying a lunar mission without adjustments. For HTV the increase in fuel would mean that it can still fit on a Falcon Heavy launch if it is expendable. Dragon could either launch on an Ariane 64 or in a dual launch option using an expendable Falcon Heavy. Cygnus would be able to fit onto an Atlas V (551) while Progress M could in principle fly using either an Atlas V (551) or an Ariane 64.

Given the available combinations of cargo spacecraft and launchers the most likely option might be an HTV+Falcon Heavy launch, which could carry 6 mt and one Cygnus+Atlas V launch which could carry the remaining 3 mt. A dual launch dragon seems not likely as the HTV with increased fuel could carry more payload. The launch costs are not considered in this analysis, as it is difficult to acquire accurate estimates for the actual launch costs of the different vehicles.

3.6. Operations Concept Methodical Approach

For the operations concept the premise that multiple ground control centers will be used is assumed. In order to optimize the distribution of functions between the centers and between the different positions within the centers a design structure matrix (DSM) based optimization approach can be used. The DSM is a systems engineering modelling method that can be used to represent the interaction between different parts of a system in a matrix. This enables mathematical methods to optimize connections between objects and activities. In this case, a team-based DSM is used. This sort of DSM is a mapping of the network of interactions among people or units within an organization [9, p. 80, Chapter 4].

	Function 1	Function 2	Function 3	⋮
Function 1	X	1	0	…
Function 2	1	X	2	…
Function 3	2	0	X	…
…	⋮	⋮	⋮	⋱

Fig. 13. Exemplary DSM for the optimization

The matrix is read from “left to top” which means that the row is representative for the element that receives an input from the elements in the columns. Otherwise, a “view from above” reveals the outputs of an element in that column. A binary mark like an “x”, “1” or similar simply shows input and output interactions. Using non-binary marks like numbers or different symbols that are representing different values results in a more detailed representation of output/input interactions, which is not always possible or necessary. For this work, non-binary (numerical) entries are necessary because of the high interlacement of the system. In this case, the elements on the left side of the rows and on top of the columns are international space station flight controller (ISS FC) core-tasks. An ISS FC core-task is defined as a task that is either done by an ISS FC that is periodically on console and/or by an ISS FC whose function is listed as core system. The challenge of using a team-based DSM for this work is to gather the necessary number of reliable datasets. A team-based DSM must work with subjective rating tools like questionnaires, which is also the method chosen for this work. A questionnaire is the chosen method here because it is necessary to use the everyday experience of a flight control team member as a base for how to optimize a flight controller’s everyday job. The DLR flight control team was the target group for this survey. More information on the survey and DSM approach will be available in Ref. [33] soon. A disadvantage of using a survey for filling in the DSM elements is the requirement of nearly equal amount of answers for every task row. This is because the clustering algorithm directly uses the element's values to calculate the optimal cluster setup. Clustering Algorithms are a form of partitioning analysis for object- and team-based DSMs. The Intention is to find subsets of DSM elements (called clusters or modules) that can be mutually exclusive or minimally interacting subsets. Because of their high interconnectedness in this case, the goal is to find groups of elements that are interconnected among themselves as intense as possible while being less connected to the rest of the system. This can be achieved by reordering the DSM's rows and columns, which is referred to as "clustering". The element's values in this work are averaged over every answer. Though, the difference in precision between certain rows leads to a result that has to be observed in question of the belonging of less precise rows to clusters with more precise ones.

The results of the survey and analysis of possible task clusters for an operations concept are currently in work and will be part of a future paper.

4. Discussion

As this analysis is currently missing some vital aspects, the selected orbit and architecture cannot be considered optimal yet. For example, Ref. [32] suggests that a round trip transfer to L2 Halo orbits with gravity assists is feasible with only 637 m/s which shows that these options must be considered. This is planned for a future analysis using the same approach as in this paper with more orbits and additional possible transfers. In addition, while the currently selected Near Rectilinear Orbit is part of the considered Halo orbits, for the L1 case no possible transfer is currently included in the analysis and for the L2 case the current Δv requirements are too high to be feasible. Therefore, a direct comparison of the currently selected orbit and the orbits considered in this analysis is not possible.

In addition, no changes to the cargo spacecrafts to make them capable of flights past LEO where considered. For example, radiation hardened electronics might be necessary. Also not considered are the possible burn durations for the different engines, these might not fit the specific mission requirements.

5. Conclusions

A possible optimization approach to select a cis-lunar orbit for a manned space station was developed. While it currently is missing some necessary data to arrive at a final conclusion the principle capability to optimize the orbit was shown. In addition, an optimized ECLSS architecture for the considered mission scenario was proposed and its required resupply masses calculated to ~4.5 kg/day for 4 crew members. The optimized ECLSS consists of a Solid Amine Water Desorption System for CO_2 removal, a Bosch reactor for CO_2 reprocessing, Static Feed Water Electrolysis for O_2 generation and a Thermoelectric Integrated Membrane Evaporation System for water recovery.

References

[1] ESA, Angelic Halo Orbit Chosen For Humankind's First Lunar Outpost. [Online] Available: http://www.esa.int/Our_Activities/Operations/Angelic_halo_orbit_chosen_for_humankind_s_first_lunar_outpost. Accessed on: Sep. 10 2019.

[2] Jason Crusan, Future Human Exploration Planning: Lunar Orbital Platform-Gateway and Science Workshop Findings,, NASA, Sep. 2019. [Online] Available: https://www.nasa.gov/sites/default/files/atoms/files/20180327-crusan-nac-heoc-v8.pdf.

[3] Jason Crusan and Robyn Gates, Cislunar Habitation & Environmental Control & Life Support Systems,, NASA, Mar. 2017. [Online] Available: https://www.nasa.gov/sites/default/files/atoms/files/20170329-nacheoc-crusan-gatens-hab-eclss-v5b.pdf. Accessed on: Sep. 10 2019.

[4] B. Schreck, Feasibility Analysis of a Life Support Architecture for an Interplanetary Transport Ship,. Master Thesis, Technische Universität München, 2017.

[5] D. Pütz and B. Schreck, Combining Multi Criteria and ESM Analysis in the Life Support Trade Off Tool (LiSTOT) to Analyze the Interplanetary Transport Ship Concept of SpaceX, in 48th International Conference on Environmental Systems, Albuquerque, 2018.

[6] A. Feigel, Advancement of a Trade-off Tool for Life Support Technologies and its Application in Proposing a Life Support Architecture for the Gateway,. Master Thesis, Institute of Astronautics, Technical University of Munich, Munich, 2019.

[7] AGI, System Tool Kit: AGI, 2019.

[8] The MathWorks Inc, MATLAB R2019a. Natick, Massachusetts: The MathWorks Inc., 2019.

[9] S. D. Eppinger and T. R. Browning, Design structure matrix methods and applications. Cambridge, Mass: MIT Press, 2012.

[10] M. K. Ewert, J. L. Broyan, K. E. Goodliff, M. Clowdsley, and R. Singleterry, Comparing Trash Disposal and Reuse Options for Deep Space Gateway and Mars Misisons, in AIAA SPACE and Astronautics Forum and Exposition, 2017, p. 5126.

[11] M. S. Anderson, M. K. Ewert, J. F. Keener, and S. A. Wagner, Life Support Baseline Values and Assumptions Document, NASA/TP-2015-218570/REV1, Jan. 2018.

[12] B. E. Duffield, Advanced life support (ALS) technologies list version 4,, *Lockheed Martin Space Mission Systems & Services*, 2001.
[13] M. McKellar, C. Stoots, M. Sohal, L. Mulloth, B. Luna, and M. Abney, The Concept and Analytical Investigation of CO2 and Steam Co-Electrolysis for Resource Utilization in Space Exploration, in 40th International Conference on Environmental Systems, 2010, p. 6273.
[14] Northrop Grumman, Cygnus™ NG-10 Mission: Cargo Delivery Spacecraft for the International Space Station,. [Online] Available: https://www.northropgrumman.com/Capabilities/Antares/Documents/NG-10_Factsheet.pdf. Accessed on: Sep. 23 2019.
[15] E. Seedhouse, SpaceX's Dragon: America's Next Generation Spacecraft, 1st ed. Cham, Heidelberg, New York, Dordrecht, London, Chichester: Springer; Praxis Publishing, 2016.
[16] Japan Aerospace Exploration Agency, HTV-1 Mission Press Kit,, JAXA, Sep. 2009.
[17] NASA, ORION QUICK FACTS,, NASA. [Online] Available: https://www.nasa.gov/sites/default/files/atoms/files/fs-2014-08-004-jsc-orion_quickfacts-web.pdf. Accessed on: Sep. 19 2019.
[18] ESA, Artemis 1. [Online] Available: https://www.esa.int/Our_Activities/Human_and_Robotic_Exploration/Orion/Artemis_1. Accessed on: Sep. 19 2019.
[19] J. P. Gutkowski, T. F. Dawn, and R. M. Jedrey, Evolution of Orion mission design for exploration mission 1 and 2,, 2016.
[20] Mark Wade, Progress M. [Online] Available: http://www.astronautix.com/p/progressm.html. Accessed on: Sep. 23 2019.
[21] O. J. Haidn, Advanced rocket engines,, *Advances on propulsion technology for high-speed aircraft*, vol. 1, 6-1, https://www.researchgate.net/profile/Oskar_Haidn/publication/225022069_Advanced_Rocket_Engines/links/54dcdde40cf25b09b912d795.pdf, 2008.
[22] Mike Carney and Eric Haddox, Launch Vehicle Performance Website. [Online] Available: https://elvperf.ksc.nasa.gov/.
[23] NASA, SLS June 2018 Fact Sheet, FS-2018-08-084-MSFC, 2018. [Online] Available: https://www.nasa.gov/sites/default/files/atoms/files/0080_sls_fact_sheet_10092018.pdf. Accessed on: Sep. 19 2019.
[24] Ariane Group, ArianeGroup to study a Moon mission for ESA,, Jan. 2019. [Online] Available: https://www.ariane.group/wp-content/uploads/2019/01/Moon-contract-2101.pdf. Accessed on: Sep. 24 2019.
[25] J. S. Parker and R. L. Anderson, Low-energy lunar trajectory design. Hoboken, New Jersey: Wiley, 2014.
[26] R. W. Farquhar and A. A. Kamel, Quasi-periodic orbits about the translunar libration point,, *Celestial Mechanics*, vol. 7, no. 4, pp. 458–473, 1973.
[27] J. V. Breakwell and J. V. Brown, The Halo family of 3-dimensional periodic orbits in the Earth-Moon restricted 3-body problem,, *Celestial Mechanics*, vol. 20, no. 4, pp. 389–404, 1979.
[28] K. Connor Howell, Three-dimensional, periodic, ?halo? orbits,, *Celestial Mechanics*, vol. 32, no. 1, pp. 53–71, 1984.
[29] U. Walter, Astronautics: The Physics of Space Flight, 3rd ed. Cham: Springer International Publishing, 2018.
[30] J. S. Parker and G. H. Born, Direct lunar halo orbit transfers,, *J of Astronaut Sci*, vol. 56, no. 4, pp. 441–476, 2008.
[31] Andreas Garhammer, Calculation of Halor Orbits and Transfer Trajectories for the Earth-Moon System,. Term Paper, Institute of Astronautics, Technical University of Munich, Munich, 2019.
[32] W. D. Pratt, C. Buxton, S. Hall, J. B. Hopkins, and A. Scott, Trajectory Design Considerations for Human Missions to Explore the Lunar Farside From the Earth-Moon Lagrange Point EM-L2, in AIAA SPACE 2013 Conference and Exposition, San Diego, CA, 09102013, p. 306.
[33] Niklas Lindig, Interface Optimization for the Gateway Mission Control Center: Work in Progress,. Term Paper, Institute of Astronautics, Technical University of Munich, Munich, 2019.