École Polytechnique fédérale de Lausanne

Minor Project in Space Technologies

EL3 Polar Explorer: Radio Antenna Payload Pre-phase A Study

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Contents

1 Abbreviations

Abbreviation	Meaning
ADC	Analogic-Digital Converter
ADPM	Antenna Deployment and Pointing Mechanism
CFRP	Carbon Fiber Reinforced Polymers
COTS	Commercial Off-The-Shelf
EL3	European Large Logistic Lander
EMI	Electromagnetic Interference
ESA	European Space Agency
ESTRACK	European Space TRACKing
LDE	Landing Descent Element
LNA	Low Noise Amplifier
NA	Not Applicable
PCDU	Power Conditioning and Distribution Unit
PPE	Payload Platform Element
RFC	Regenerative Fuel Cells
RLS	Robotic Landing Stack
RTG	Radioisotope Thermoelectric Generator
SA	Solar array
SADM	Solar Array Drive Mechanism
SNR	Signal to Noise Ratio
TBC/D	To Be Confirmed/Determined

2 Applicable and reference documents

Ref	Title
Statement of Work	EL3 POLAR EXPLORER RADIO ANTENNA PAYLOAD PRE-PHASE A STUDY- EXPRO+
AD1	Lunar Surface Radio Antenna: Preliminary Science Requirements
AD2	PRELIMINARY SYSTEM REQUIREMENT DOCUMENT FOR EL3 Pre Phase A STUDIES
RD1	Planetary and exoplanetary low frequency radio observations from the Moon
RD2	L-DEPP Executive Summary Report
RD4	EL3 Polar Explorer: Science Model Payload Definition Document (MPDD)
RD5	Jupiter's low-frequency radio spectrum from Cassini/Radio and Plasma Wave Science (RPWS) absolute flux density measurements
RD6	ELVIS – ELectromagnetic Vector Information Sensor
RD7	The Netherlands-China Low Frequency Explorer (NCLE)

Table 1: Applicable and reference documents

3 Introduction and problem definition

3.1 Scope of the project

This project aims at proposing a payload concept for the " European Large Logistic Lander (EL3) polar explorer ", the first European Spatial Agency (ESA) technology demonstration mission of the EL3 mission. This mission, planned to be launched in 2028, aims at preparing sustainable human exploration activities on the Moon. One of the various considered payload for this first mission is a radio antenna. This project will focus on this payload.

3.2 Objectives of the mission

The objective of this mission is "to demonstrate the suitability of the lunar surface as a platform for low-frequency radio astronomy and to try and provide a first measurement of the long wavelength radio emission (2 to 60 MHz) [...]" [Statement of Work]. These wavelengths correspond to the highly red-shifted HI emissions. As discovered by Edwin Hubble in 1929, the expansion of the universe is observed through a red-shift of the light of galaxies getting further from us. Hubble described the speed of the expansion by a simple law, stating that the further the object is, the faster it moves away from us. For that reason, and because light travels at a finite celerity, highly red-shifted wavelengths corresponds to the furthest objects and at the state the universe was several billion years ago (the " cosmic dark age"). It is thus scientifically crucial to observe these wavelengths. The solar system's planet's low radio emissions are also a source of interest [RD1]. However, these observations can not be done with Earth based telescopes due to Earth's ionosphere being opaque to these wavelength. Man-made low frequency radio signals interfering with the desired signals also prevent good observation. In consequence, observations on the surface of the Moon would solve most of these issues since the Moon's ionosphere is significantly less dense than Earth's. In addition to that, putting a radio antenna on the far side of the Moon would allow the Moon's body to shield the antenna from Earth Radio Frequency Interference (RFI) [RD1]. Polar regions are also considered for this application because although less shielded, these regions have more frequent sunlight leading to easier operations (available solar power, temperature) and easier communication with Earth. These regions are also where man is planned to set foot on the next lunar human expeditions. This mission aims at showing such observations can be made. Characterize the lunar environment and identify low radio signals sources on the lunar surface is critical to the success of future observations as other sources of radio signal are present (sun radio emissions, galactic background radiation, quasi-thermal and photon-electron electrostatic noises) [RD1]. The longer term goal for lunar surface low frequency radio astronomy is interferometry, a technique in which an array of antennas is used, effectively acting as a giant diameter telescope, thus greatly increasing its resolution.

4 Review and Requirements definition

In order to fulfill the goals of the mission, the science requirements have to be satisfied. These requirements, written by ESA, are driven by the science goals of the mission. They are listed in [AD1] and are recalled below.

'Shall' indicates requirements that must be accounted for and met.
'Should' indicates requirements for which solutions should be proposed and then accepted or excluded based on an assessment of the resultant technical risks and feasibility.

Identifier	High-level Requirement	Measurement / Parameter; Notes
RA-RQ-010	The payload shall be capable of determining the broad band radio environment including in the low-frequency range.	~80 KHz to 100 MHz range.
RA-RQ-020	They payload shall measure all three components of the electric field. *To attain the 3D polarisations of radio signals; to estimate direction of approach (DOA) of signals, and to aid with discriminating interfering signals.*	Tripole or three-monopole antenna configuration [RD1] (TBD). Cross-correlation between antenna inputs is required.
RA-RQ-030	The payload shall characterise and monitor the radio background (and its sources) on the Moon in the spectral and temporal domain.	Temporal scales: integration times of milliseconds to seconds. Monitoring timescale: continuous. Spectral resolution: sub-kHz.
RA-RQ-040	The payloads shall enable the detection of radio reflection and transmission of the Moon surface (e.g. crater rims, mountains) and regolith below 100 MHz (i.e., dielectric constant).	
RA-RQ-050	The payload shall conduct experiments to measure the signal from the cosmological dark ages (HI at redshift z=30-200).	From 7 to 100MHz
RA-RQ-060	The payload shall conduct experiments to measure the signal from the cosmic dawn (HI at redshift z=15-30)	
RA-RQ-070	The payload shall be able to capture the dynamical timescales of radio sources including: a) on the order of hours/days/weeks (e.g. emission of the large planets, the Sun, AKR and other radio noise sources) b) on the order of extreme short timescale events (e.g. lunar cosmic ray impacts, and Jupiter and solar flares)	Lower limit is nano-second voltage-traces, upper limit is weeks.
RA-RQ-080	The payload shall be able to achieve scientific objectives despite electromagnetic interference from the lander.	E.g. capability for antenna to be mounted on a boom, electromagnetic shielding of payload and platform components.
RA-RQ-090	The payload shall have a minimum lunar surface operational lifetime of 2 years.	
RA-RQ-100	The payload shall be capable of being deployed robotically or by humans.	

Figure 1: Science requirements table taken from [AD1]

In order for the science requirements to be satisfied, top-level functionalities of the payload can be determined. From these functionalities, requirements can be written to describe these functionalities in measurable ways. Later in the process, a functional analysis of the payload will be made where lower level functionalities of the payload can be determined.

4.1 Top level functionalities

Table 2 lists top-level functionalities of the payload. They are briefly justified in the right column. However, these functions will be decomposed in lower level functionalities later which will allow to understand them better.

Function	Explanation
Place measuring device on lunar surface	The radio antenna must be placed on the lunar surface, with the desired position, orientation and such a way that it is shielded from the differences sources of perturbation.
Measure radio signals	The received signals must be acquired, amplified and treated in such a way that they can be stored, sent and later analysed for scientific purpose
Communicate with Earth	The measurements need to be transmitted to Earth in order to be analysed by scientists. In addition to that, it might be needed to send other data such as housekeeping data. Depending on the mission scenario/ autonomy of the system, it could be necessary to send command to the lander/antenna.
Regulate temperature	Actively or passively regulate the temperature of the temperature sensitive elements during different phases of the mission and modes of operations
Generate electrical power	The different subsystems will need to be powered. This power can be used to process data, send the data, regulate temperature, etc

Table 2: Top-level functionalities

4.2 Top-level system requirements

In order to meet the different functionalities listed in table 2, preliminary requirements can be drawn. They are listed in tables 3- 6 along with justifications, details, measurable parameters or comments.

4.2.1 General requirements

Identifier	Requirement	Measurement/Parameter/Notes
RQ-GEN-010	The payload mass shall not exceed 1500 kg	The Lander descent element (LDE) and the Payload Platform Element (PPE) have a combined mass inferior to 1600 kg. The total mass must thus be 3100 kg.
RQ-GEN-020	The Payload and the Lander (LDE + PPE) shall fit in an Ariane 64 fairing	Dimensions are a diameter of 4.5m and up to 6m in height
RQ-GEN-030	The landing area shall suit the EL3 capabilities	slope $< 15°$ and rocks < 50cm and outside of a shadow zone

Table 3: Top-level general system requirements coming from EL3 and launcher capabilities

4.2.2 Functional requirements

Identifier	Requirement	Measurement/Parameter/Notes
RQ-FUN-010	The system shall include a receiving antenna and the associated hardware necessary to measure and treat radio signals	see the preliminary science requirements in annex 1
RQ-FUN-020	The system shall place and/or shield the measuring device on the lunar surface such that it is shielded from the different sources of electromagnetic perturbations	Earth man-made radio signals, solar perturbations and the payload's and the platform's components
RQ-FUN-030	The system shall deploy the antenna in the desired position and orientation	Position and orientation TBD
RQ-FUN-040	The system shall be able to communicate scientific and housekeeping data to Earth	quantity of data TBD
RQ-FUN-050	The system shall be able to receive command from Earth	quantity of data TBD
RQ-FUN-060	The system shall generate, store, control and distribute electrical power according to the needs of the different subsystems during the different operating modes of the payload	-

Table 4: Top-level functional system requirements

4.2.3 Environment requirements

Identifier	Requirement	Measurement/Parameter/Notes
RQ-ENV-010	The system shall withstand and properly function under lunar temperature during all phases of the mission	The temperatures include the lunar days and night. These temperatures depend on the landing site. Note that different functioning mode can be designed for days and night.
RQ-ENV-020	The system shall withstand and properly function under the lunar radiation environment	13.2 $\pm 1\mu$ Gy/hour [1]
RQ-ENV-030	The system shall withstand and properly function under the lunar dust environment	

Table 5: Top-level environment system requirements

4.2.4 Reliability requirements

Identifier	Requirement	Measurement/Parameter/Notes
RQ-REL-010	The system shall operate for at least 2 years.	-
RQ-REL-020	The system shall contain enough redundancies	The goal is to avoid single points of failure (SPoF) as much as possible

Table 6: Top-level reliability system requirements

5 Payload High level interfaces and design

5.1 Functional analysis

In order to meet the science requirements of the mission, the system Lander + Payload will need to perform certain functions. The following tree shows the top-level functions the system will need to perform as well as a decomposition into sub-functions for the most complex ones.

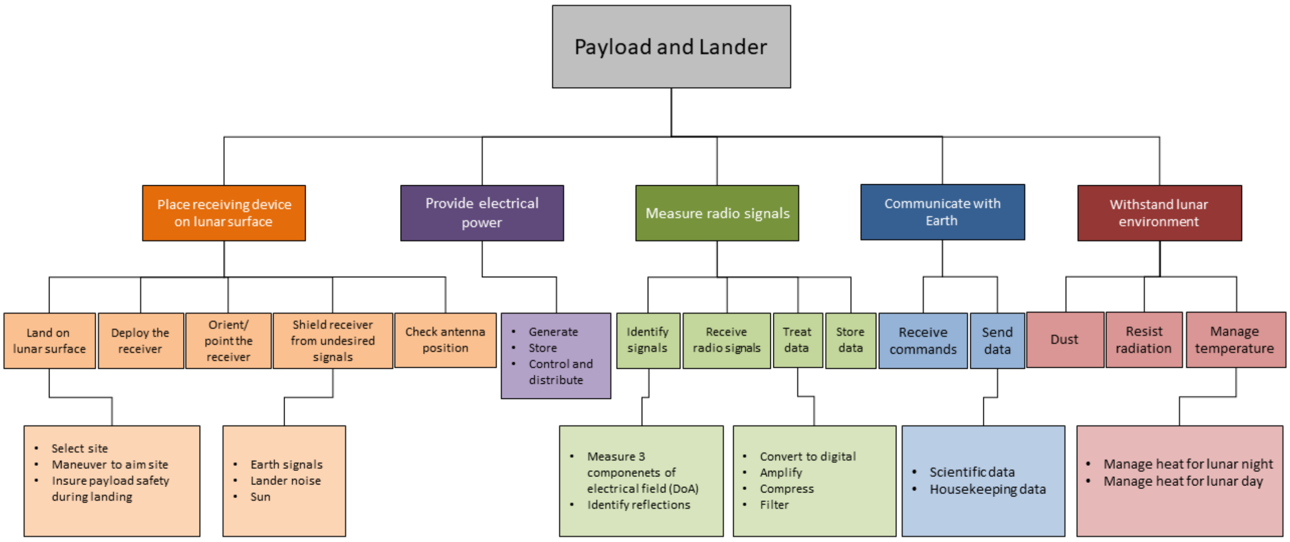

Figure 2: Functional decomposition tree for the Lander-Payload system

5.2 Morphological matrix

For each of the functions that need to be performed, several solutions/ physical components can be thought of. Listing them in a morphological matrix is a good way to visualize all the possibilities. At this step, no idea is rejected even if some components seem unpractical, not realistic or even crazy. Several possible systems can then be represented as paths on this matrix.

Importance [1-4]	Function	Sub-function	Solution 1	Solution 2	Solution 3	Solution 4	Solution 5	Solution 6
4	Land on Lunar surface	Select site	pre-selected exact site	Camera + OBC + algo	Humans decide when deploying			No precise selection
4		Maneuver to aim sit	EL3					
4		brake to land	EL3					
4	Deploy the receiver		Electrical motors + boom	Pyrotechnic	Pneumatic	Spring loaded	Antenna on rover	No deployment
4	Orient the receiver		Electrical motors		Pneumatic		Antenna on rover	No pointing
4	Shield the receiver from perturbations	Earth Signals	Go on hidden face	Go at bottom of a crater?				
4		Lander noise	Boom	Roll at a distance	shoot antenna at a distance	EM Shielding		Don't
2		Sun	Observe during lunar night	Place antenna in deep crater				Don't
1	Check antenna position							
4	Generate electrical power		Solar panels	Fuel cells		RTG or equivalent		
4	Store energy		Battery	regenerative fuel cells	Mechanically (inertia or potential)			Don't store energy
4	Control and distribute power		PCU					
4	Receive radio waves		Tripole antenna	three monopole antenna	antenna dish			
4	Identify signals origin	Measure 3 compone	Three orthogonal arms for DOA		steer antenna dish			
4		Identify reflections	camera to identify surrounding topology	radar	lidar	on orbit reconaissance	With DOA only?	
4	Treat data	Amplify	LNA					
?		Filter	filters					
4		Convert to digital	ADC converter					
?		Compress	OBC					
4	Store data		Solid state mass memory	MRAM	SRAM	DRAM		no storing
2	Receive command from Earth		Low gain antenna	Same antenna as scientific data				no command needed
4	Send data to Earth	Scientific	Low gain antenna	Relay satellite	To LOPG	directly to earth	physically take memory device	
3		Housekeeping	Low gain antenna	Same antenna as scientific data				
3	Withstand Lunar dust		Shield with dust proof material	Stay off ground				
3	Withstand radiation		Shield electronics with alu	Redundant electronics				
4	Manage Temperature	Hot	MLI	paints	heat pipes	radiators		
4		Cold	MLI	paints	heat pipes	radioisotop heaters (RHU?)	electrical heaters	cogeneration of fuel cells

Figure 3: Morphological matrix for the payload

With no loss of generality, a product tree can be made. Note that at this point, no choice of component is made and the tree should be solution dependant. This tree will help define terms later use in the mission concept generation phase.

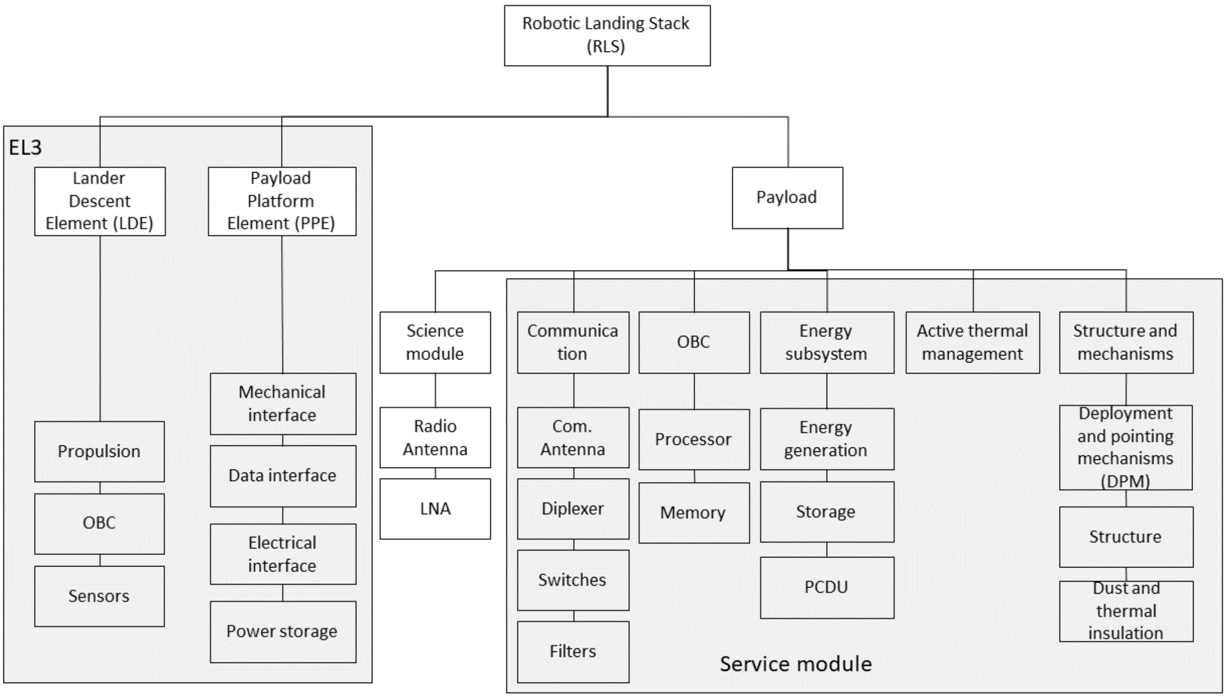

Figure 4: Product tree

5.3 Mission Concepts generation and trade-off

The goal of this section is to generate a few mission concepts. For each mission, landing site and the concept of operation are main drivers for hardware choice. For each concepts, without going into too much detail, choices will be made in order in order to answer the following questions:

- Where will the EL3 land?
- Will the payload separate from the lander?
- Will the payload have mobility?
- How will the science module deploy?
- How will power be generated and stored?
- How will the system survive lunar nights (54 of them)?
- How will communication with Earth be achieved?

5.3.1 Concept generation

Two main options can be chosen regarding landing site. While landing on the pole offers significant advantages regarding power generation, thermal management and communication with Earth (possibility of almost continuous daylight and direct line of sight with Earth), the far side offers greater science measurement conditions because the Moon's body shields the antenna from Earth made signals as well as the sun during Lunar night. These advantages for data collection are important drawbacks regarding power generation, thermal management and communication with Earth.

After the landing, the payload can stay on the lander, can be deployed next to it or can even be imagined having the mobility in order to go further away from the lander. This mobility could also be used to choose a specific observing site.

The following diagram summarizes the main options that can be thought of.

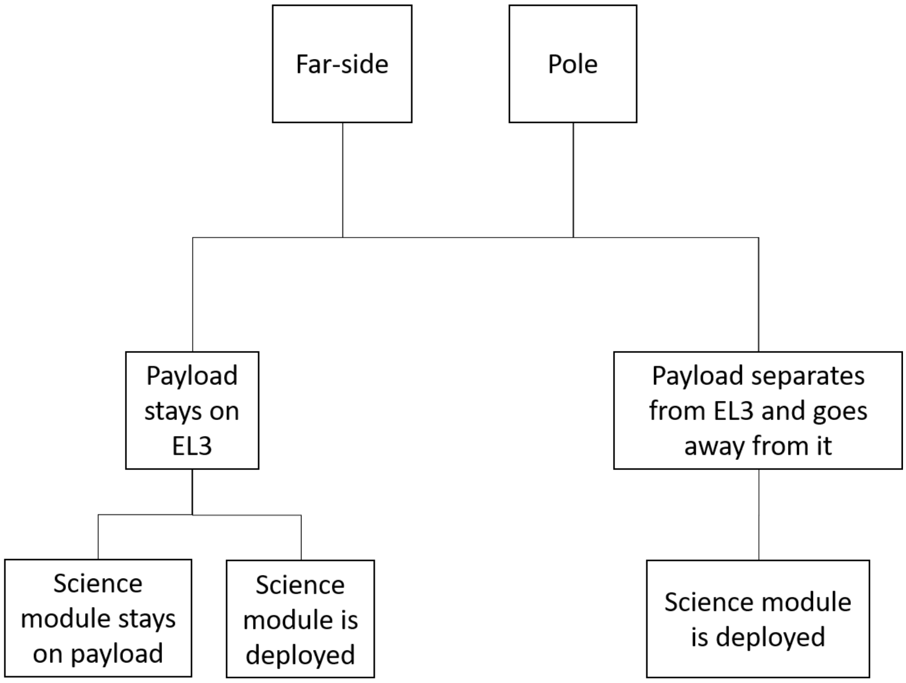

Figure 5: General mission concept generation tree

Note that for each general choice, several choices can then be made. For example, when chosen to go at the pole, landing on the rim of a crater can be chosen in order to have almost continuous sunlight, it can also be chosen to go at the bottom of the crater where observation conditions can be better. The option of landing on a crater's rim and putting the payload at the bottom can also be studied although it seems hard to achieve. In addition to technical difficulty, it is not clear that the bottom of the craters are as well shielded as the far side from Sun and Earth perturbations. When choosing a mobile system going away from the lander, it can be chosen to be completely autonomous from the lander allowing it to go far away or can use subsystems present on the lander via a tether to remain light, containing only subsystems essential to the science measurements. This way, the subsystems causing perturbations can stay far from the Science Module.

5.3.2 Concept 1

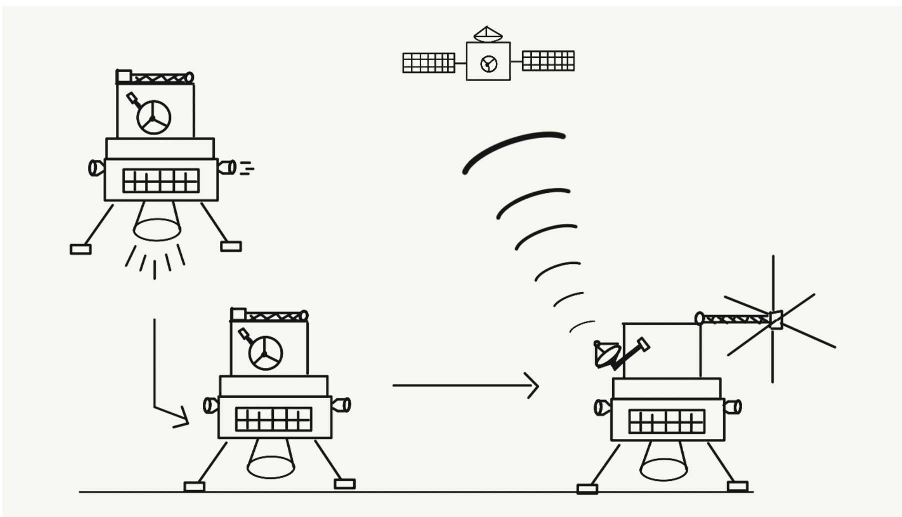

Figure 6: Concept 1

The first concept is a simple case. After landing on the far-side, the whole Robotic Landing Stack (RLS = LDE + PPE + Payload, see Figure 4) stays together (the payload has no mobility). Some components of the LDE, PPE and payload can thus be mutualized. The elements that need to be are deployed (scientific payloads, communication antenna and eventually solar arrays). On the far-side, interesting measurement can be made during night and day, allowing to characterize solar perturbations during the day and to have less perturbed measurements during the night (there are other sources of noise that remain at night [RD1]). Communication with Earth can be made using the Lunar Gateway as a relay. In order to function, the system can use a combination of solar arrays and regenerative fuel cells (RFC). The latter technology, still in development would allow storing energy for the long lunar nights. Another option to be considered would be Americium-241 RTGs that are currently being developed by ESA.

5.3.3 Concept 2

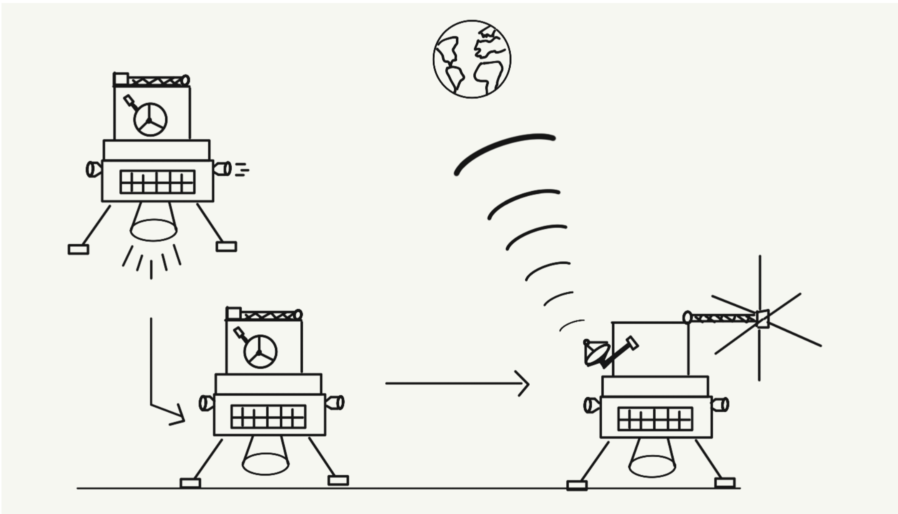

Figure 7: Concept 2

The second concept is also simple. After landing on the south pole (exact location TBD), just as the first concept, the whole RLS stays together and the different elements are deployed. The difference is that because depending on the chosen landing site, the lunar nights can be much shorter and Earth can be in direct line of sight, power storage can be much simpler, batteries can probably be used (battery technology is much more advanced than RFCs) and communication can be made with Earth directly (via ESTRACK for example). These technical simplification however probably imply that the low frequency radio measurements will be less shielded from Earth and the Sun.

5.3.4 Concept 3

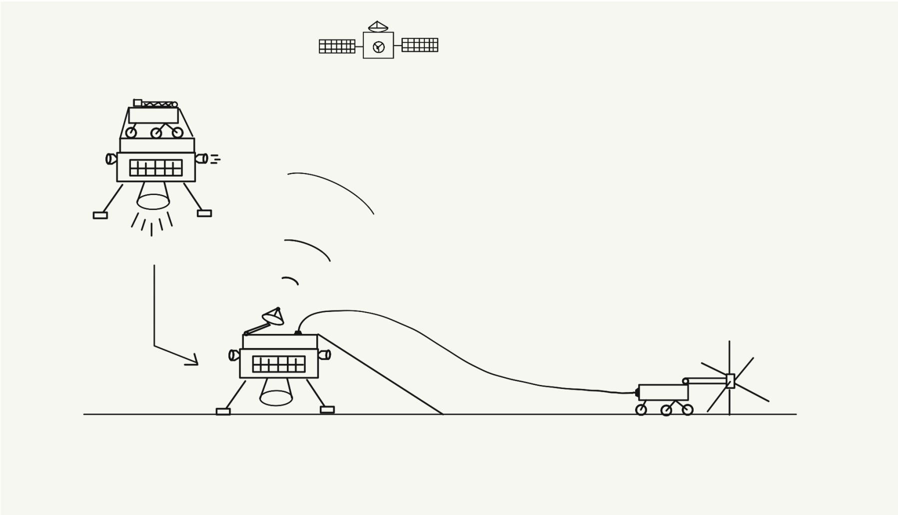

Figure 8: Concept 3

The third concept consists in a mobile science module. First, the RLS lands on the far side. By going far from the lander (10's to 100's of meters), the payload will be less subject to the Electromagnetic Interferences (EMI) generated by the other subsystems (remaining of the payload, PPE and LDE). Tethering the mobile science module to the remaining of the RLS allows to isolate only the essential (antenna, LNA, ADC and some heat management components) from the RLS. The other functions can then be performed on the LDE + PPE stack, power and data being transmitted through the tether. Power generation and communication can be performed just as in concept 1 (SA + RFCs).

5.3.5 Concept 4

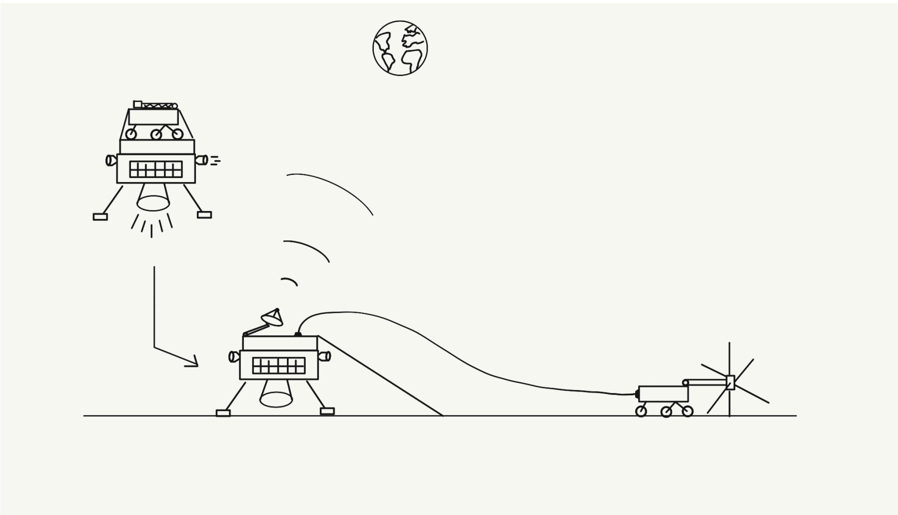

Figure 9: Concept 4

Concept 4 is similar to the previous but the landing site is located at the pole. Thus, it communicates and generates power just as in concept 2. This concept would be more interesting if the mobile payload is capable of going to a more shielded area such as inside of a crater. It is unclear that measuring conditions inside crater are really better than outside of it since the observed wavelengths are very long. However, it is quite probable that it is less favorable than on the far-side during the night.

5.3.6 Concept 5

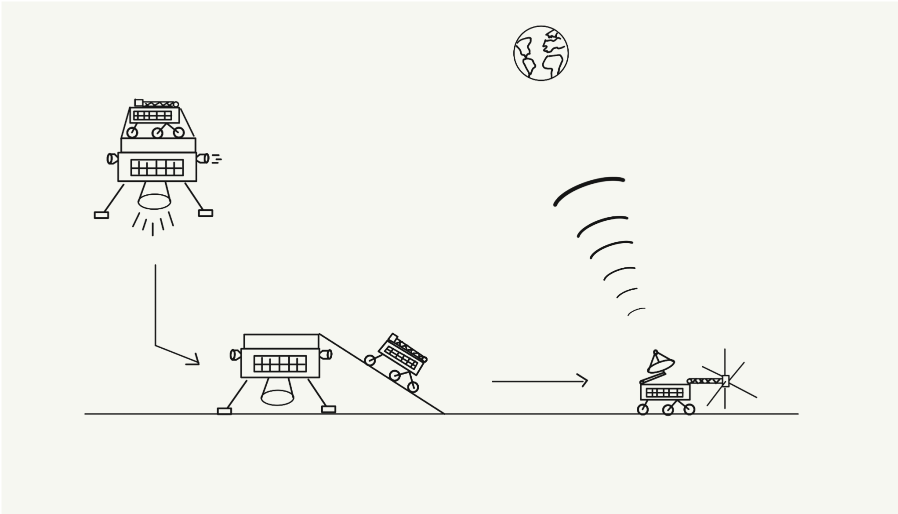

Figure 10: Concept 5

Concept 5 goal is to use the mobility of the payload to make measurements at different locations. In this case, the mobile payload is autonomous from the lander and can thus travel far more distance (1 - 10's of km). In a first phase, the payload could thus make measurements on the crater rim, then descend into the shadows of a crater and then finish its energy reserves making measurements there. The problem of this concept is first that it is quite complex and will have a very limited life time inside the crater. Secondly, the inside of the craters are probably not as well shielded as the far-side concerning radio signals coming from the Earth and the Sun. Note that ESA plans to develop rovers capable of driving bigger distances than what is currently possible. Such a rover is described in [2] and could be adapted to carry radio antennas. One explored concept (PHILIP) is to power the rover from the RLS with a laser. This would allow the rover to stay in the darkness of a crater while remaining powered at a distance by the RLS which has access to a lot of sunlight on the crater rim.

5.3.7 Summary

	Site	Mobile	Tether	Power gen.	Storage	Comm.
C1	Far-side	no	NA	SA or RTG	RFC or nothing	Gateway
C2	Pole	no	NA	SA	Batteries or RFC	Gateway or Earth
C3	Pole (rim)	yes	no	SA	Batteries	Gateway or Earth
C4	Far-side	yes	yes	SA or RTG	RFC or nothing	Gateway
C5	Pole (rim)	yes	yes	SA	Batteries or RFC	Gateway or Earth

Table 7: Summary of the different mission concepts

Notes:

- Deploying a static science module next to the lander with a robotic arm was considered but doesn't seem useful. Letting the science module on a boom would be simpler, offer similar shielding from the lander and better shielding from lunar dust.

- Having an autonomous mobile payload is not very efficient for shielding the antennas from the RLS EMI because some subsystems will be present on the LDE+PPE and on the payload, thus weighing more. In addition to that, the mobile system would necessarily bring sources of EMI along with the antenna.

- The justification for choosing an observation site not accessible by the lander and thus the need for a rover would have to be very strong as it is not expected that observations conditions vary a lot on distances scales of hundreds of meters to a few kilometers (distances that rovers are able to travel).

5.3.8 Choice of concept

The far side is the most well suited observation site for several reasons. First, the alternating of lunar nights and days allow measurements not only during these periods but also at dawn or dusk. This will help characterize the effects of the sun on low radio frequency observations while always staying shielded from man-made radio signals coming from Earth. There, it can use the whole body of the Moon while being at the pole (in direct line of sight of the Earth or hidden inside a crater) would not provide this shielding (or not enough).

The communication there is less straightforward as a relay is needed. However, this is still achievable and at the time the mission is set to be launched, the lunar gateway will be able to fulfill this role. Temperature wise, the site is better than at bottom of a crater, temperatures are well characterized and thermal management system can be designed accordingly thanks to a lot of potential electrical power available during the day.

Deploying science module within the lander is the most effective strategy because it is simple and the deployment of antennas is very well mastered. Having the science payload move away from the RLS on a rover would, in addition to bringing more complexity, not improve the measurements in significant ways as sources of noise would be brought with the science module. Indeed, theses sources of noise come from components that would need to be brought along the science payload if the latter was to travel long distances. A tethered rover would also bring a lot of complexity for a shielding that could be provided more simply. Indeed, noise from other components can be mitigated by putting the radio antennas on booms, sufficiently far from the RLS and limiting the activities of the non-essential components during radio observations. In addition to that, electromagnetic shielding can be designed to protect the science module from the lander EMI. These reasons led to the choice of the first concept.

5.4 Payload design

5.4.1 Science module

As determined in the L-DEPP study [RD2], a set of orthogonal (or quasi orthogonal) dipoles are well suited for lunar based low radio frequency measurements. As described in [RD1], using two or three dipoles allows " to derive instantaneously the 4 Stokes parameters, i.e. the intensity (S) and full polarization (Q,U,V) [...] together with the two angles (θ, ϕ) defining the direction of the wave vector **k**". Using 2 or 3 dipoles also allows to have a SNR larger by a factor of $\sqrt{2}$ or 2 to that obtained with a single dipole.

The critical drivers for the antenna design are linked to their geometry, how they will fit on the RLS in the stowed position, how much volume they take and how they can be deployed. Another important design driver is their geometry (section and length) and composition as this will determine the deflection they will have under lunar gravity. Note that having an antenna on the Moons surface adds this complexity. Indeed, for an antenna staying in space, the antennas can be made flexible and stowed on a reel as in [RD6] however such flexible antennas would not remain straight under lunar gravity.

The deflection of an antenna can be written as: $\delta_{max} = \frac{g_{Moon}}{8} \cdot \frac{\rho}{E} \cdot \frac{A}{I} \cdot l^4$
where $g_{Moon} = 0.166 \cdot g_{Earth}$, ρ and E are respectively the material's density and Young's modulus, A and I the area and the quadratic moment of the arm section and l is the length of the arm.
While $\frac{\rho}{E}$, the inverse of stiffness to weight ratio (or specific modulus) is relatively similar for common used metal alloys (aluminum, steel, titanium), it is significantly better for CFRP. Concerning $\frac{A}{I}$ which is purely a function of the section, it is way better for a tube ($\frac{4}{R^2+r^2}$) than for a cylinder ($\frac{4}{R^2}$), especially as the tube becomes thinner.

As an example, Figure 11 shows deflection of antenna arms as a function of length for cylindrical and tubular geometries.

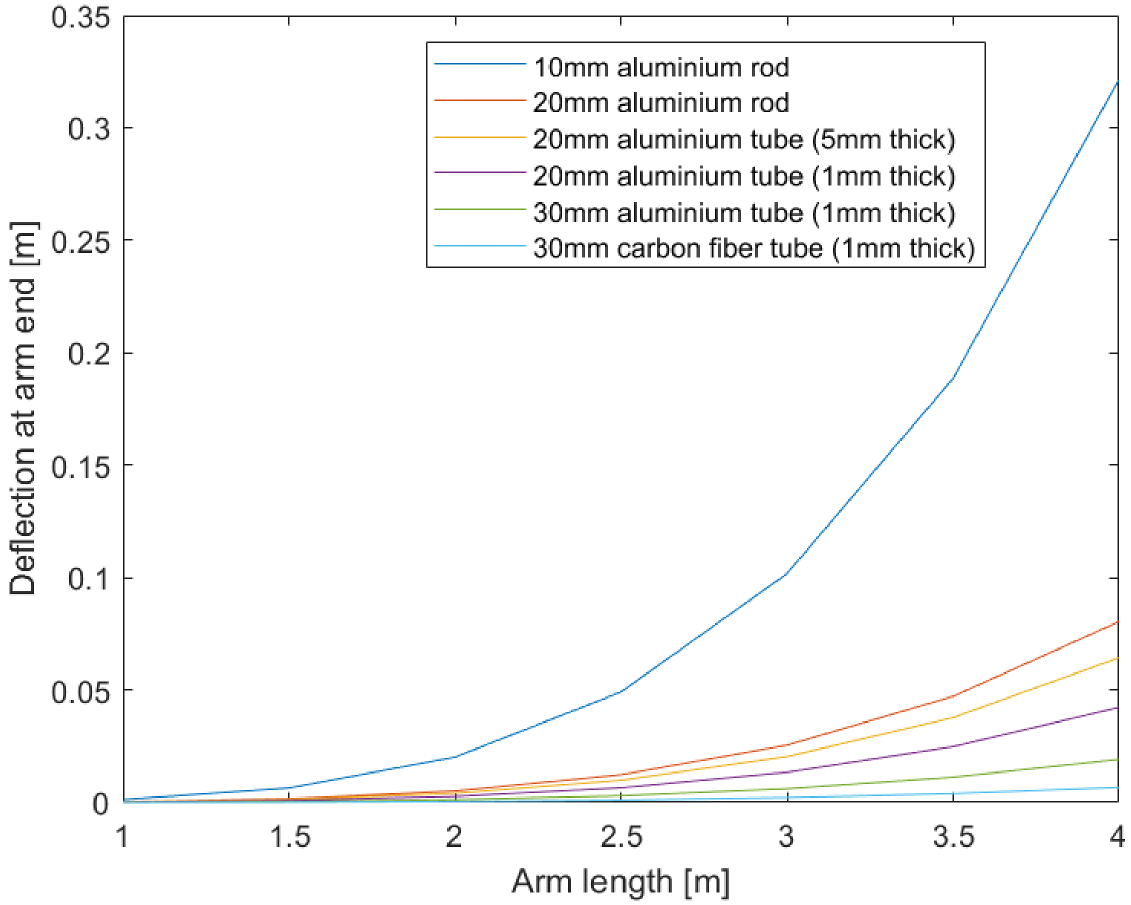

Figure 11: Deflection of antenna arms as a function of length for cylindrical and tubular geometries

For these reasons, choosing CFRP tubes as antenna seems the best way to minimize deflection under lunar gravity as the length of the antennas increase. It is especially important as this deflection increases as l^4. The conducting element can be chosen as metallic foil covering the tube or a wire. An option that can be studied is to use the carbon fiber itself as the conductor. This as been done and studied in [3] where carbon fiber in graphene containing binder has been used as a conductor. The investigation, although for a different frequency band (200-600 MHz) showed potential performances comparable to metallic antennas. This option can be studied although it is less mature and is more complex to realize (especially the electrical connections).

The general approximate dimensions are shown bellow. These dimensions set constraints on the size of the antenna when stowed in order to fit on the RLS, when deployed in order not to touch the ground and be at sufficient distance from the lander (avoiding EMI) and during deployment to avoid interference.

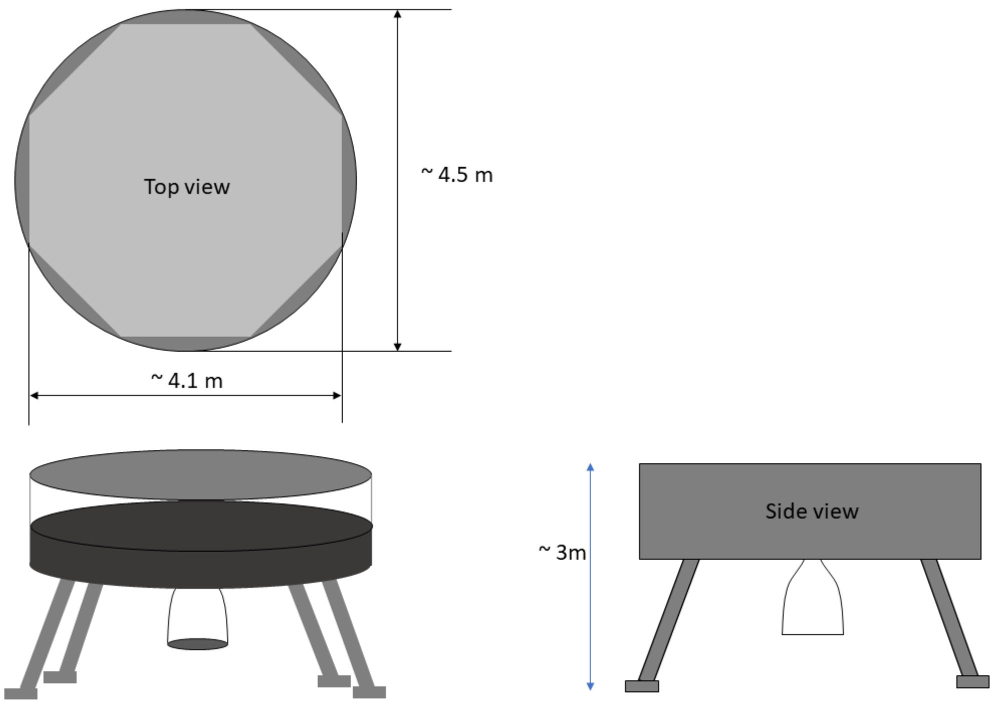

Figure 12: General dimensions of the RLS taken from [2]

From these constraints, three configuration concepts are drawn.

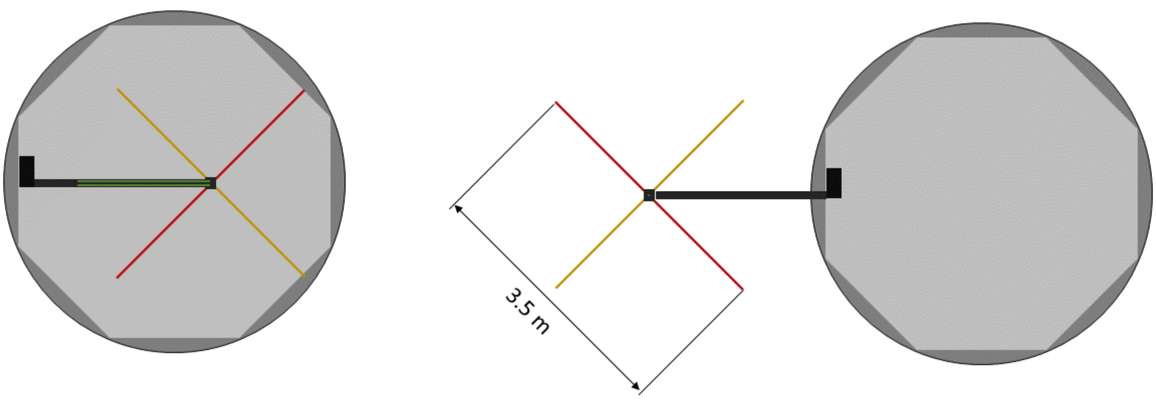

Figure 13: First configuration (approximately at scale)

In the first concept (Figure 13), the three dipoles are located on the same boom. Two actuators are used to deploy and orient the boom. In order to fit on a plane, one of the dipoles is stowed and deployed thanks to a third actuator. The latter can be situated at the base of the boom and deploy the stowed dipole thanks to a cable and pulley system as shown on Figure 14 and 14. Putting the dipole deployment actuator at the base of the boom allows to reduce the torque needed for boom deployment, it also reduces the deflection of the boom under lunar gravity. The tension in the cable can even be used to reduce this deflection of the boom, thus allowing for a smaller diameter boom allowing for an overall lighter system.

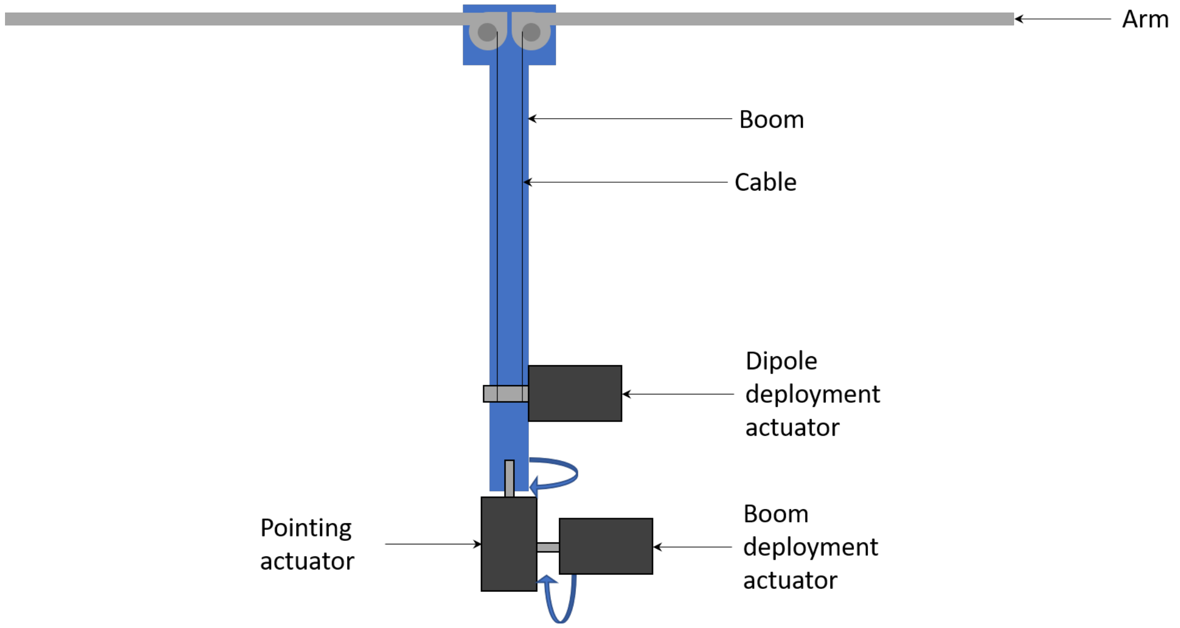

Figure 14: Possible dipole deployment mechanism

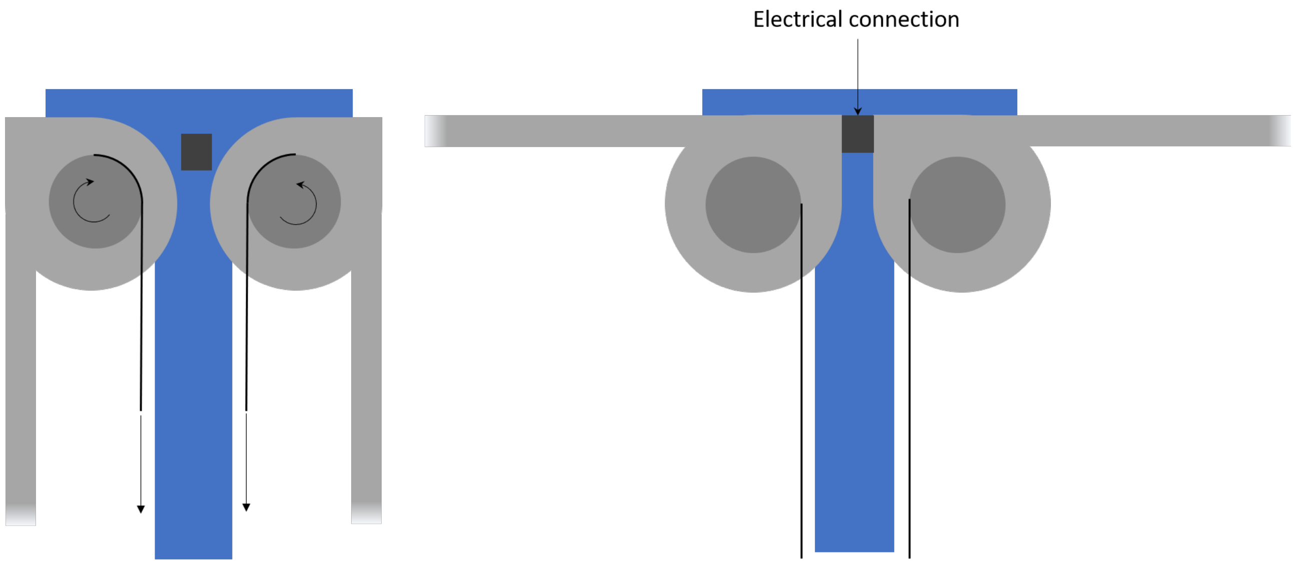

Figure 15: Possible dipole deployment mechanism

In order to maximize the possible arm length, the second concept (Figure 16) consists in the separation of the three dipoles. That way each arm can be as long as the diameter of the PPE minus the space occupied by the deployment and pointing mechanism. In this case, each dipole is deployed and pointed as shown previously on Figure 14 and 15

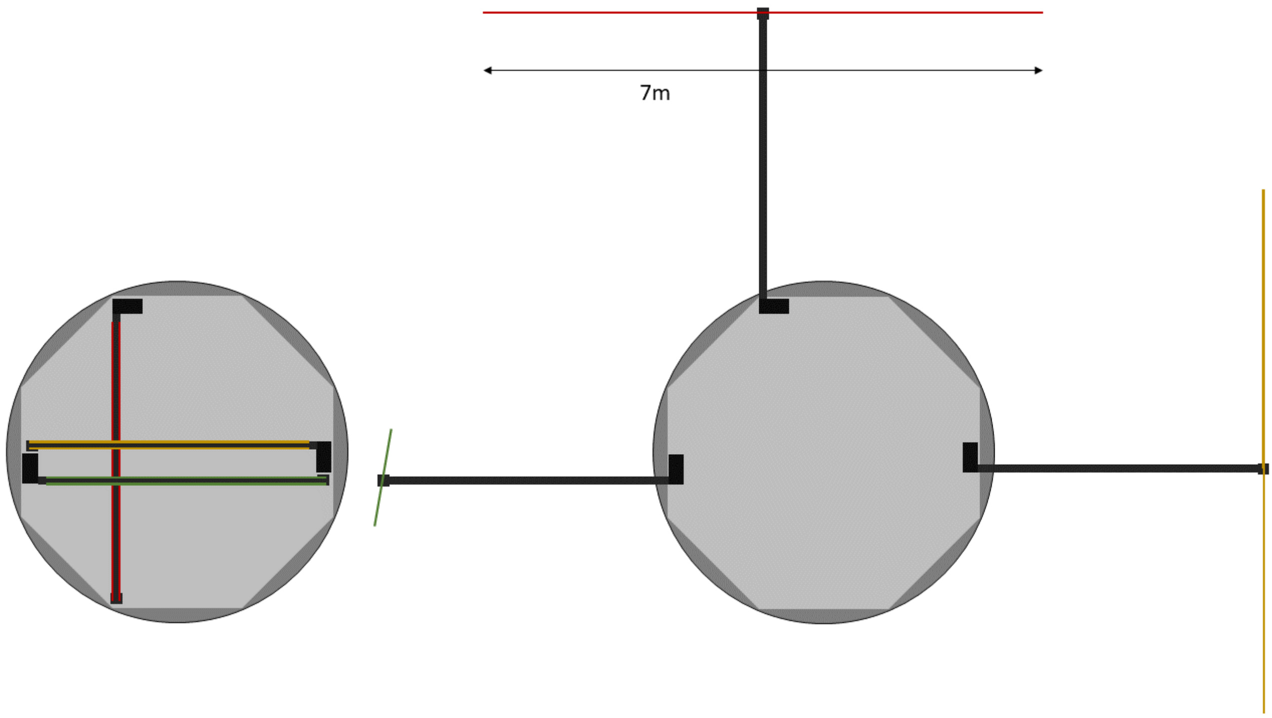

Figure 16: Second configuration (approximately at scale)

The third configuration is quite similar as the second but by shortening the arms and rearranging the booms on the PPE, much more room is available for other payloads.

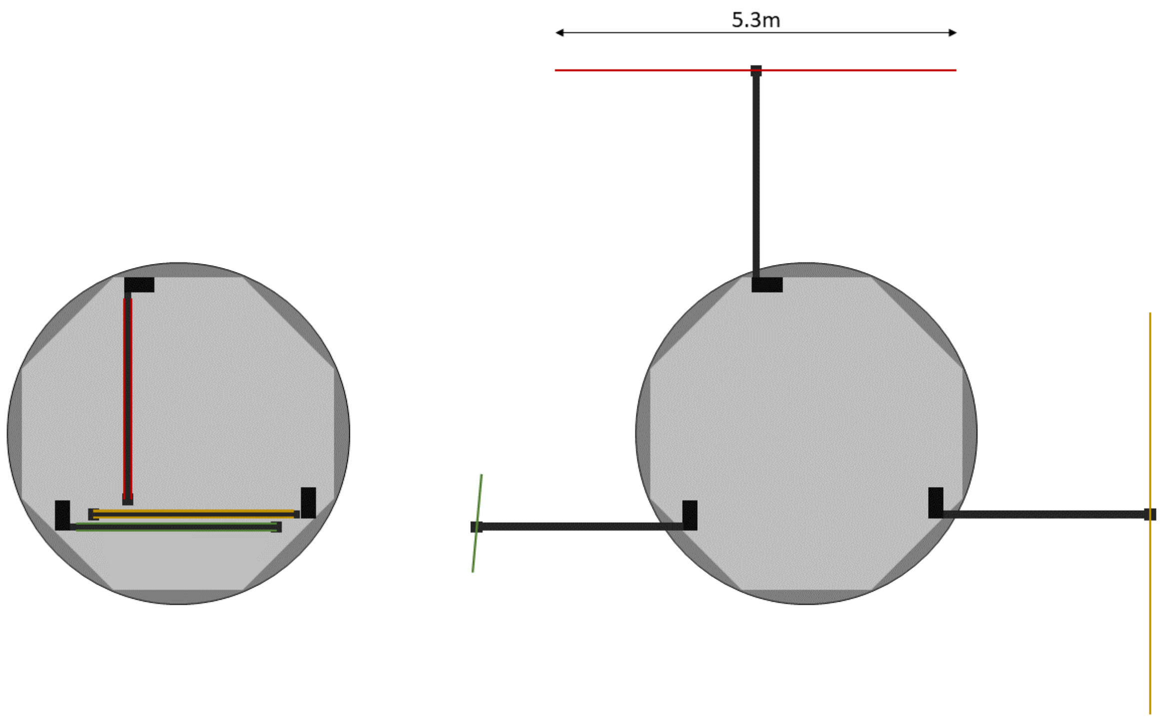

Figure 17: Third configuration (approximately at scale)

Table 8 summarizes the probabilities of successful deployment for each configuration assuming probability of any actuator properly functioning being n and values assuming $n = 0.99$ and $n = 0.999$.

	At least one dipole			At least two dipoles			All three dipoles		
	-	$n = 0.99$	$n = 0.999$	-	$n = 0.99$	$n = 0.999$	-	$n = 0.99$	$n = 0.999$
1	n	0.9900	0.9990	n	0.9900	0.9990	n^2	0.9801	0.9980
2	$2n^2 - n^4$	0.9996	0.9999	n^4	0.9606	0.996	n^6	0.9415	0.9940
3	$3n^2 - 3n^4 + n^6$	0.99999	0.99999	$3n^4 - 2n^6$	0.9988	0.99999	n^6	0.9415	0.9940

Table 8: Probabilities of partial and full deployment for the three proposed configurations

Table 9 summarizes the principal characteristics of the three configurations

	Max dipole length [m]	Total mass estimation [kg]	Boom deployment torque [Nm]	Arm deployment torque [Nm]	Room for other elements
1	3.5	10	11	0.4	- -
2	7	30	18	1.5	-
3	5.3	28	12	0.8	+

Table 9: Comparison of the three proposed configurations, see part 7.1 in annex for assumptions made in order to obtain these values

Notes:

1. For concepts 2 and 3, having booms the same size as an arm ensures clearance between the antenna and the ground or the RLS (See figure 24 in Annex)
2. Note that total mass is not really a relevant criterion because having a lot of spare mass is useless if other payloads don't fit on the RLS body. The mass Because the third concepts offer the advantages of having 3 separate independent dipoles, it has a high probability of at least partial deployment. This is interesting because although 3 dipoles is the best, goniopolarimetric measurements can still be made with two and basic science can still be achieved with one [RD1]. In addition to that, the third antenna configuration offers significant room for other elements that needs to fit on the RLS body (solar arrays, communication antennas, radiators, other payloads, etc).
3. In order to increase maximum dipole length, further folding each arm in two or more parts can be be a possibility although adding complexity and mass. In addition, clearances with the ground and distance with the RLS would have to be ensured. If these clearances are not met, the boom itself could be folded or be telescopic to increase in length once deployed.

5.5 Power needs estimation

In order to plan the configurations of other elements on the RLS, a power estimation has to be done. 4 different modes can be distinguished.

Deployment Mode : just after landing, the different elements deploy (communication antennas, radio antennas, other science payloads, potentially solar arrays). Because each actuator can be functioning one after the other and science measurements are not made during this mode, it is a mode that doesn't require a lot of power. It can be assumed that this mode functions thanks to the energy storage.

Day Mode (DM) : During day mode, the system can make day measurements which will help characterizing the environment during the day. This mode is very power consuming because in addition to make the system function, the power system must recharge the energy storage subsystem for the coming night. Although communication and science measurements would not be done simultaneously, they are put together in this mode as a first approximation. In this mode, heat management is taken into account in a larger contingency (30%) as it is not easy to estimate how much the heat rejection can be made passively or actively.

Night Mode (NM) : On this mode, the subsystem makes measurements and must actively heat in order to maintain the functioning temperatures of the different subsystems. It can only rely on the energy storage. It can be decided if the RLS will communicate with the Lunar Gateway or not but it will here assume it will not. Instead, it will store the data and send it during the day, when more power is available.

Night Survival Mode (NSM) : Because the NM is very energy demanding, a portion of the night can be spent in NSM. In this mode, the temperature to be kept is lower (components usually have two specified temperature ranges, one for functioning and one for survival, the latter allowing for lower temperatures). In this mode, the components that can are shut down (communication, payload) while minimum activity is kept on the others (monitoring, heating, etc). The sizing of the SA and the energy storage will thus depend on the ratio between NM and NSM.

The estimation of the power is iterative and made thanks to a Matlab code. The logic of the calculation is shown on Figure 18 while the assumptions about radiating surfaces (area and material) are directly visible in the code.

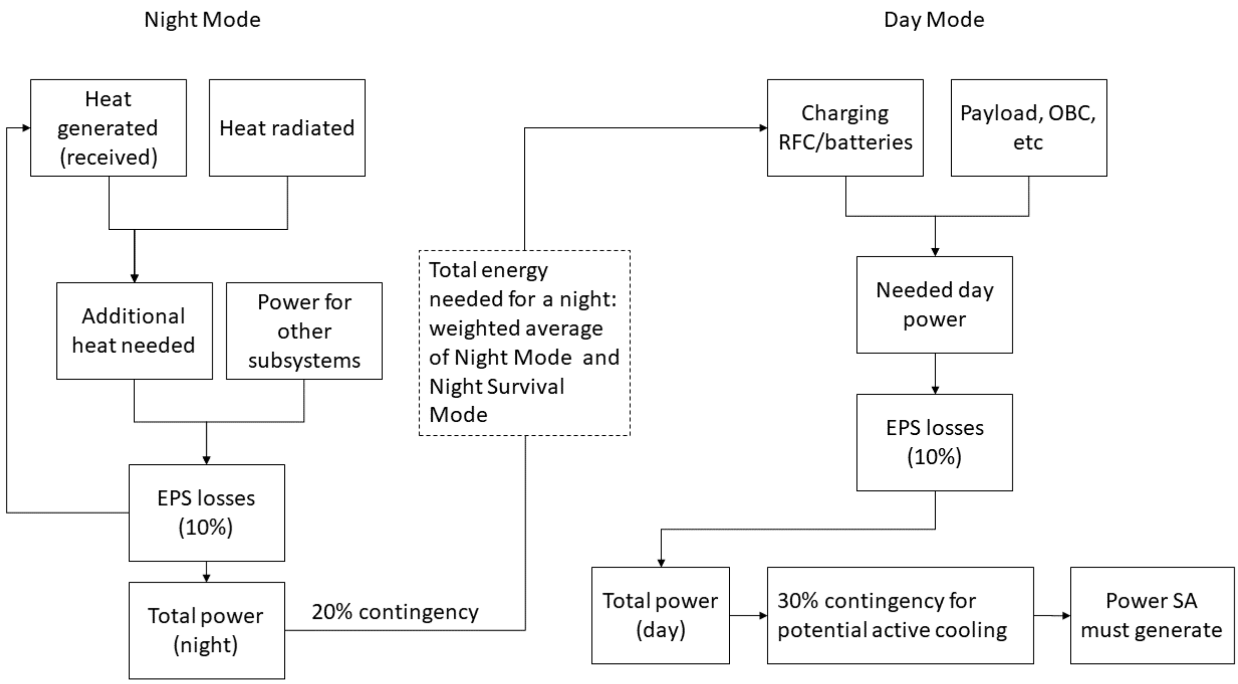

Figure 18: Method to estimate power needs

Element / Mode	Source, reference or justification [m]	Deployment Mode	DM	NM	NSM
Payload [W]	L-DEPP [RD2]	0	20	20	0
OBC [W]	SOA-SS (MOOG) [4]	25 *	25 *	25	5 **
Heat Management [W]	Calculated	***	***	125	30
Communication (transponder+ amplifier) [W]	SOA-SS [4]	20 *	20 *	0	0
Charging fuel cells [W]	Calculated	0	122	0	0
Deployment mechanisms [W]	20Nm at 1rad/s	20	0	0	0
EPS losses [%]	Calculated	10	10	10	10
Total [W]	-	108	234	189	54
Contingency [%]	-	30	30	20	20
Total + contingency	-	140 W	280 W	230 W	65 W

Table 10: Power estimation for the different modes. This power budget, although being simple and unrefined, is relatively consistent with power budget for similar concepts in [2]
*Estimated using COTS components ** Rough estimation (TBC), *** In contingency

5.5.1 Solar array surface area

Assuming a 20% solar cell efficiency, the sun being at an average 30° angle and an additional 320% contingency on the area, 2.7m ^{2} are needed to provide the necessary power during the day. During this time, ∼ 40 kWh of energy need to be stored for the night assuming 50% of the night is spent measuring and 50% is spent in survival mode.

Such a surface can be mounted onto the body of the RLS as shown on Figure 19. By choosing the location right, the useful space it occupies can be minimized (under the stowed antennas)

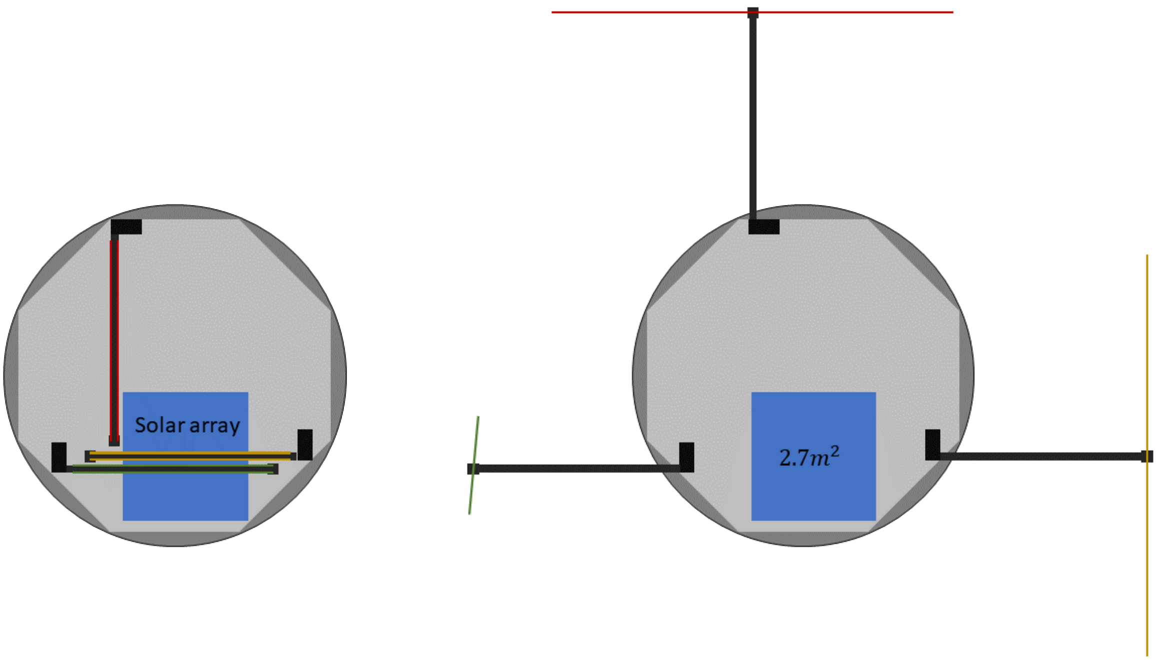

Figure 19: Solar array configuration possibility, on PPE (approximately at scale)

In order to save some space on the PPE, the SA could also be folded (shown on Figure 20) or rolled (roll-out solar arrays, see Figure 21) and deployed after landing. Along with saving some volume, it would also allow to protect from dust during landing and to orient to follow the course of the sun. This would be made at the cost of more complexity and in consequence more risk of failure. Concerning the mass, SA for small satellite use have specific power ranging anywhere from 45 W/kg to 165 W/kg [4]. This range can be explained by several factors including how they are mounted on the satellite (body mounted, deployed flexible, deployed rigid). Rigid panel solar array systems by Redwire [5] have masses ranging from 0.02 to 0.15 kg/W (specific powers from 7 to 50 W/kg). A specific power of about 50W/kg will be used here, corresponding to an low-average deployed rigid SA from [4] and a feasible technology for [5]. This results in a 5.6 kg solar array. A 100% contingency will be added to that mass to take into account the extra structure needed to support it under lunar gravity. A SADM such as the SEPTA 33 made by RUAG [6] (with an unpowered holding torque 20 Nm) has a mass of around 4.25 kg (including external leads and connectors). The SADM and structure mass will be added later to the mass budget even though a body mounted SA is still a possibility.

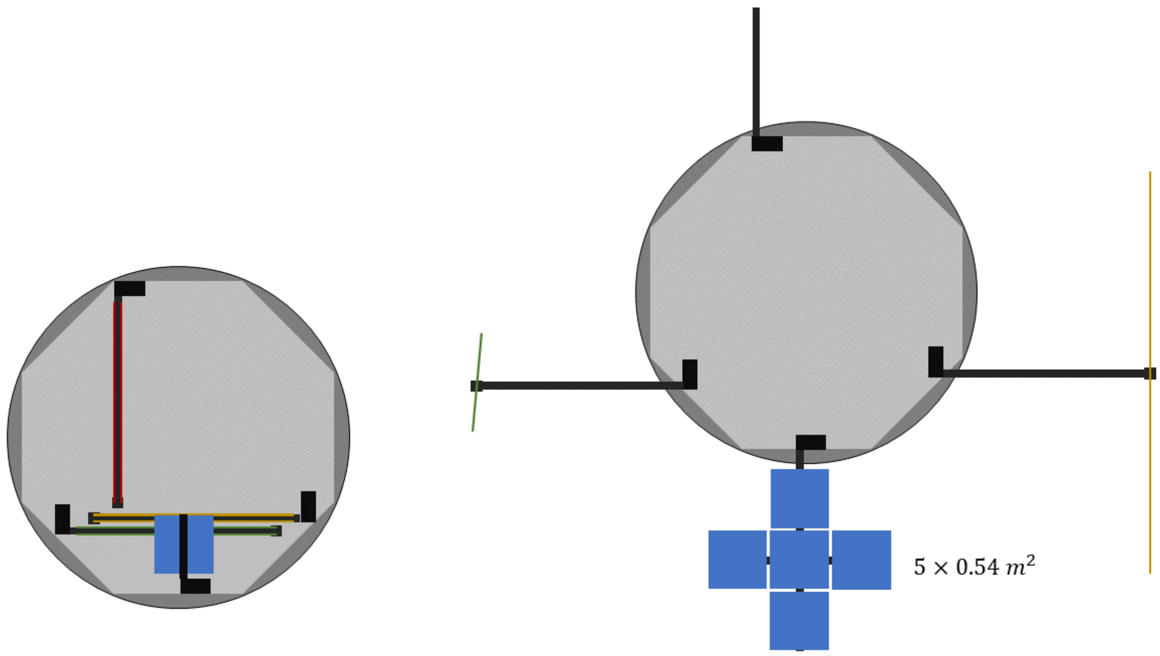

Figure 20: Solar array configuration possibility, deployed from stowed position after landing (approximately at scale)

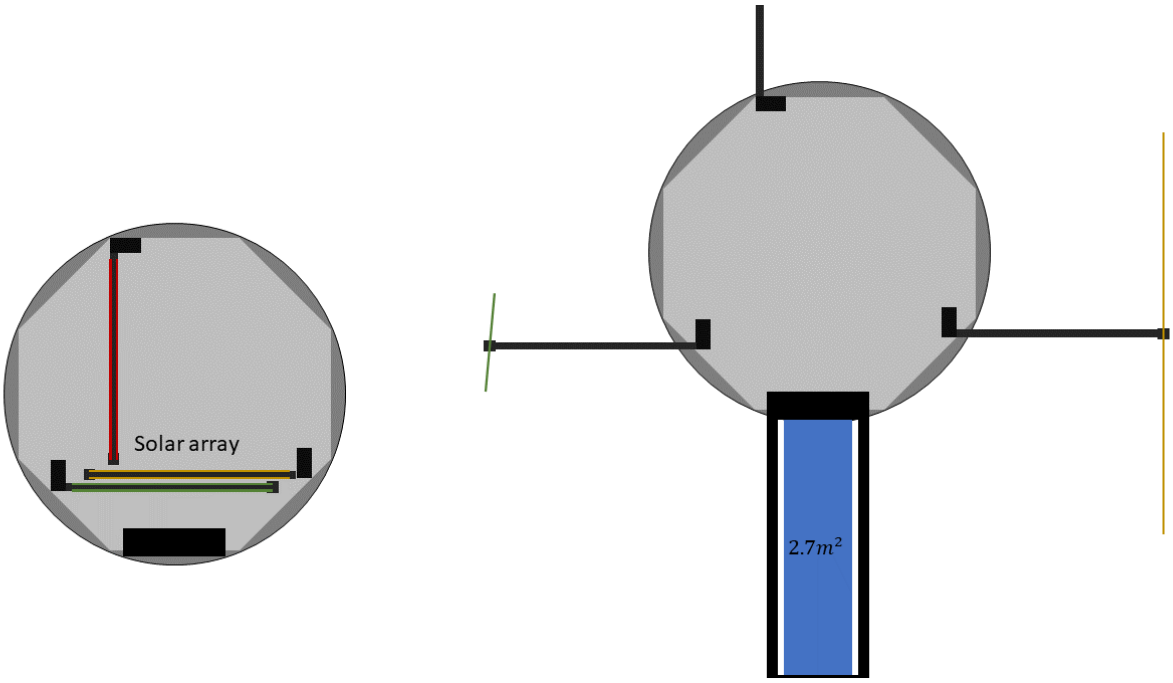

Figure 21: Roll-out SA (approximately at scale)

As seen on Figure 22, fuel cell systems are the most suited technologies for this application (discharge time of days-weeks).

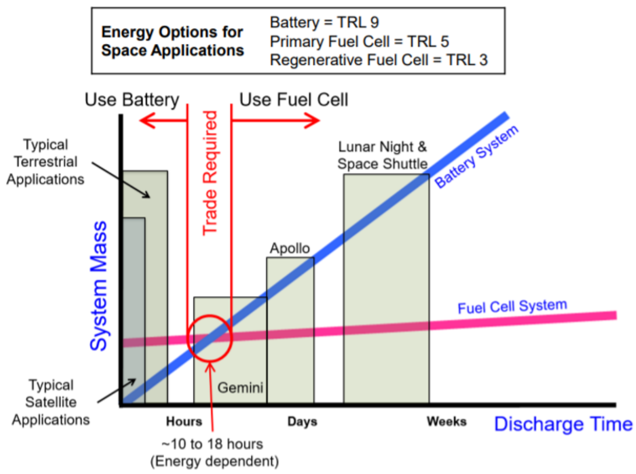

Figure 22: Lunar RFC Trade Study taken from [7]

NASA and ESA are both currently developing rechargeable fuel cells which should be operational by 2028 and offer the relatively high energy density of fuel cells while also being able to be recharged. With a typical specific energy of 200 Wh/kg [7], a battery system would have a mass of around 200kg for the needed 40kWh. On the other side, specific energy of around 500 Wh/kg could be reached by RFCs, allowing for a energy storing system of $\sim$ 80 kg (specific energy of 550Wh/kg for a 10kW H_2O_2 RFC energy storage system in [7], a large contingency (40%) will be added to the estimated mass to take into account for that difference in scale).

5.6 Preliminary mass budget

Element	Source	Unit mass [kg]	Quant.	Total mass [kg]	Subsystem contingency [%]	Total + contingency [kg]
Antenna arm	Calculated	0.4	6	2.4	20	2.88
Dipole deployment mechanism	Estimated (< an ADPM)	3	3	9	20	10.8
Boom	Calculated	0.7	3	2.1	20	2.52
Boom deployment mechanism	typical ADPM	5	3	15	20	18
LNA	L-DEPP [RD2]	0.017	3	0.05	20	0.06
OBC	[4]	5	2	10	20	12
SA	[5]	5.6	1	5.6	100	11.2
SADM	[6]	4.25	1	4.25	20	5.1
Energy storage	Calculated from [7]	80	1	80	40	112
Comm. antenna	[4]	1	2	2	40	2.8
diplexer, transponder, amplifier	Inspired by Cassini, Galileo New horizon TBC	10.6	2	21.2	20	25.44
Total	-	-	-	-	-	202.8
Total + 20% contingency	-	-	-	-	-	243.4

Table 11: Preliminary mass budget

Important note
This mass budget is very incomplete and thus not very useful at this stage. Some elements not taken into account include radiation shielding, thermal management components, insulation, harness, monitoring sensors, other EPS components and other payloads.

6 Conclusion

This report describes a system using the EL3 to land on the far-side of the moon in order to measure low frequency radio signals and characterize the lunar radio environment for future missions. In the proposed concept, the whole RLS stays as one module, three separate dipoles are deployed from the PPE on their respective boom. Each dipole is composed of two arms that are unfolded after landing. This way, the dipole length can be on the order of twice the length of the PPE. Folding each arm in two or more parts would allow for even longer dipoles although clearances with the ground and the RLS would need to be ensured. The mass and power sizing that was made in this project can be useful when looking at the order of magnitude but must be refined in future work. One of the first steps in this future work include refining the models for heat during day and night with varying parameters (such as surface, surface area, presence of active cooling, etc). Improving the model for electrical power needs and generation is also to be worked on in the future. In addition to that, evaluating a data budget and a sizing of the communication subsystem is necessary (antenna size, amplifier power, communication block diagram with redundancies, data storage, etc). Finally, choosing and sizing components will be needed in order to iterate between the different budgets, the mission requirements, the mission phases and scenarios and the addition of other payloads of interest.

7 Annex

7.1 Assumptions made in order to compare the three proposed antenna configurations

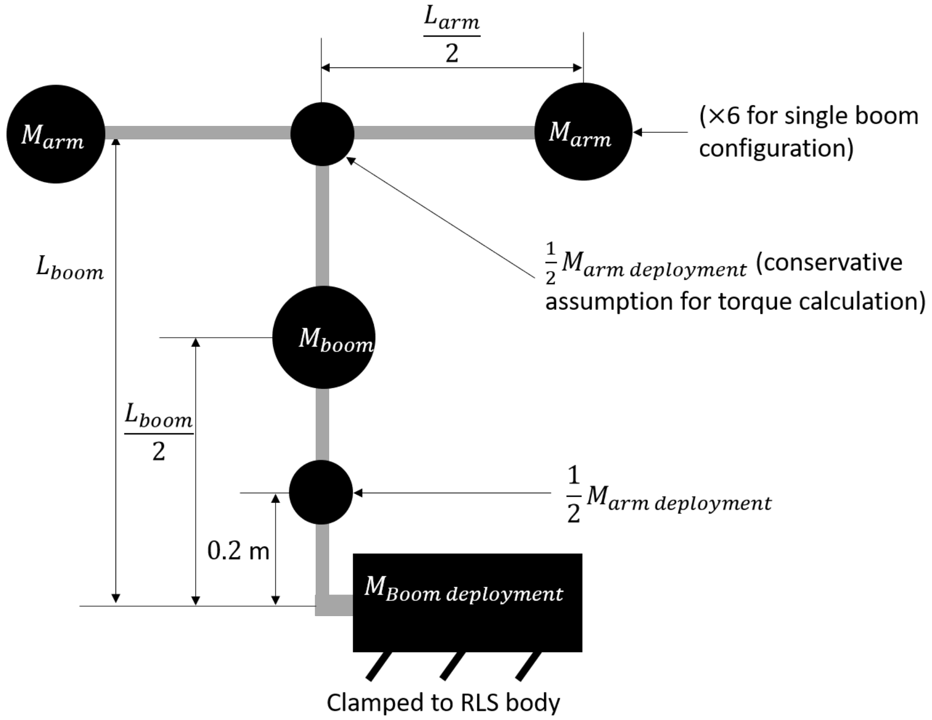

Figure 23: Mass model used to compare masses and torques of the three proposed configurations

Other assumptions:

- Mass of a boom deployment mechanism: 5 kg
- Mass of an arm deployment mechanism: 3 kg
- Arm CFRP tube, 30 mm diameter, 1mm thick (see antenna choice section)
- Boom: CFRP tube, 50mm in diameter, 50mm thick (bigger but deflection less critical, can be made up for by pointing/deployment mechanism)
- For torque: no dynamic effect taken into account, only the torque needed to work against lunar gravity is taken into account. Different effects would have to be considered for the sizing of the mechanism.

7.2 Example of deployed configuration for antenna concept 2

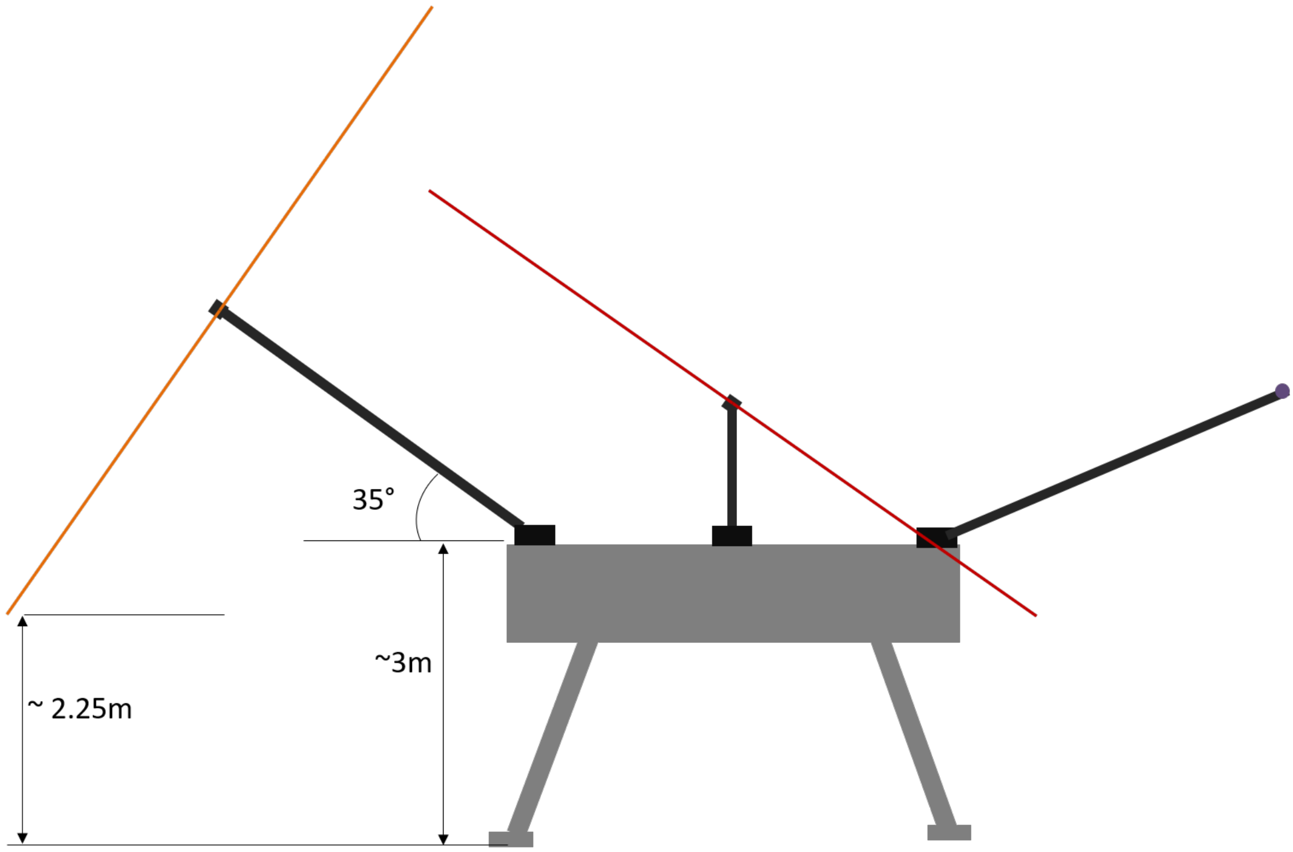

Figure 24: Deployed configuration of the second antenna concept. With booms as long as each arm, clearance with the ground is ensured for the angle with horizontal (35°) proposed by L-DEPP study in [AD1]. The distance with the RLS is equal to the boom's length. In [2], this clearance is proposed to be "1m (TBC)" for instruments sensible to EMC ("2.5m TBC" for radio antenna). These criteria are met for both concepts 2 and 3.

8 References

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