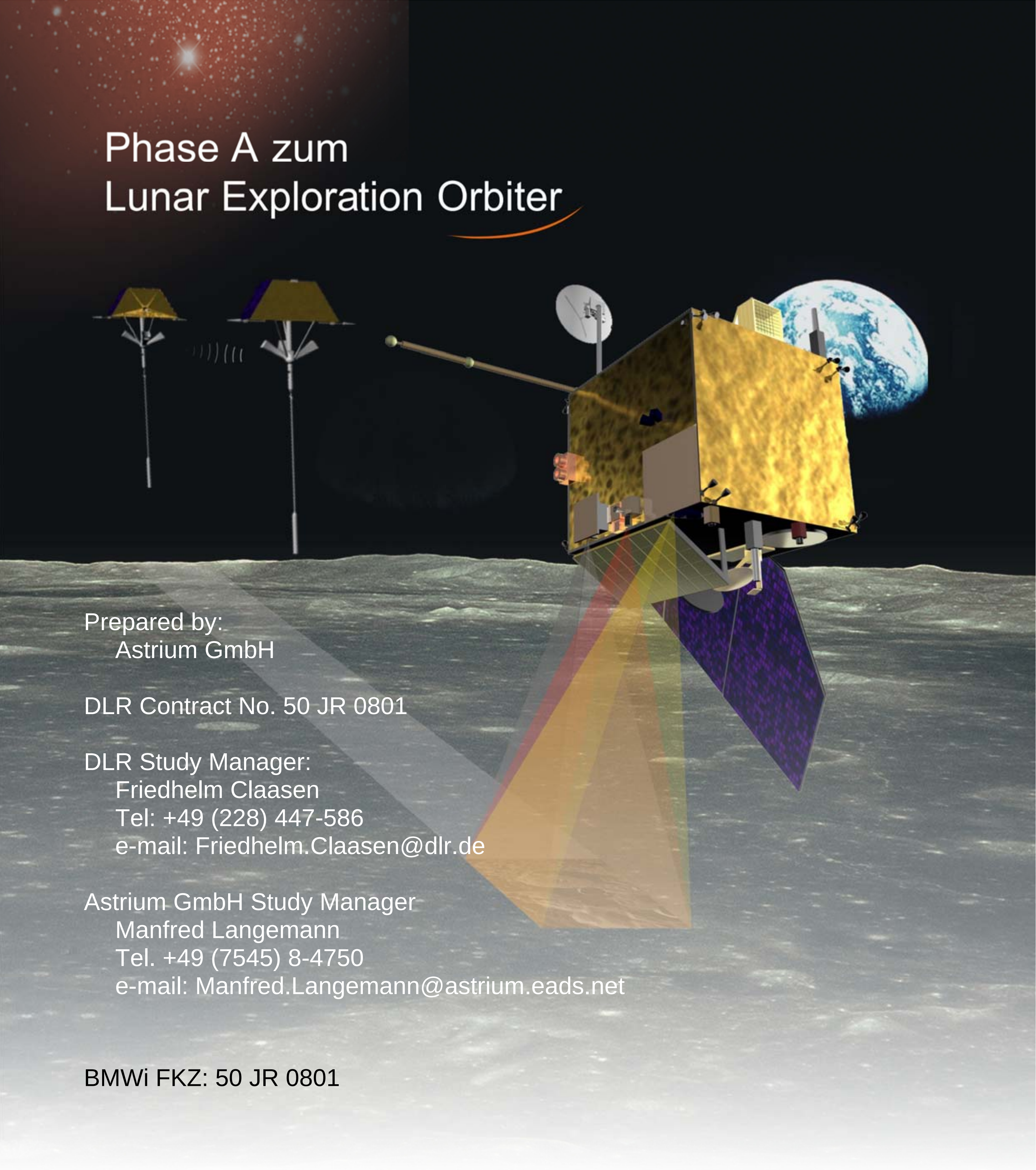

Phase A zum Lunar Exploration Orbiter

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Final Report Summary

LEO-MI-ASG-FR-003 Issue 1
30. November 2008

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Project: **Lunar Exploration Orbiter (LEO)**

Lunar Exploration Orbiter (LEO)

Phase A

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Final Report Summary

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18. Kurzfassung
Der Mond als Forschungs- und Explorationsziel ist in letzter Zeit wieder verstärkt in den internationalen Fokus gerückt. Einerseits erfordert die von den USA geplante bemannte Wiederkehr zum Mond eine genauere Untersuchung und Kartierung, anderseits bildet der Mond eine ideale Vorbereitung für die Exploration unseres Planetensystems und hält viele Antworten auf grundlegende und aktuelle Fragestellungen zum Erde-Mond System parat. Ein Beispiel unter vielen ist die Abschätzung des Gefahrenpotenzials von Meteoriteneinschlägen für die Erde. Die heute zur Verfügung stehenden Technologien erlauben eine umfassendere und um Größenordnungen genauere Erkundung des Mondes im Vergleich zu früheren Missionen.
Die primäre Zielsetzung der Phase A Studie war die Identifizierung und Konsolidierung der wissenschaftlichen Missionsziele, des Missionskonzepts sowie der Instrumentkonzepte. Hierbei dienten die Ergebnisse der vorausgehenden Phase 0 Studien als Ausgangsposition.
In der Phase A wurde die Machbarkeit der Mission auf nationaler Ebene nachgewiesen.
Das Missionskonzept sieht einen Direkteinschuss in die lunare Transferbahn von 3-4 Tagen Dauer mit einer Sojuz vor. Am Mond werden ein Hauptsatellit und zwei Tochtersatelliten 4 Jahre in einem 50 km Orbit mit einer polaren Inklination von 85° verbleiben. Auf dem Hauptsatelliten sind 12 Instrumente, im Wesentlichen zur Kartierung der Mondoberfläche, vorgesehen, während die Tochtersatelliten mit der PRARE-L Instrumentierung ein Experiment zur Schwerefeldvermessung bilden und noch zwei weitere Instrumente tragen.
Die deutsche Wissenschaft und Industrie ist sehr gut für weitergehende nationale und internationale Mond- und Planetenmissionen aufgestellt.

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18. Abstract
In the present time the exploration of the moon has an international revival. The plans of the USA for a return of the human to the moon require detailed analysis and mapping of the moon. On the other hand, the moon is not only an ideal springboard for further exploration of our solar system but also is key to answering fundamental and current questions on the Earth-Moon system. The assessment of the potential thread to Earth by impacts of meteoroids is one example.
State-of-the-art technology allows for more comprehensive and orders of magnitude more precise exploration of the moon than previous missions.
The primary objective of the phase A study was the identification and consolidation of the scientific mission objectives, of the mission concept and of the instrument concepts. The results of the preceding phase 0 studies have been the basis for these investigations.
The feasibility of the mission on national level has been shown in the phase A.
The mission concept foresees a direct injection with a Sojuz launcher into the lunar transfer orbit, which lasts about 3-4 days. A main satellite and two sub-satellites will remain 4 years in low lunar orbit of 50 km altitude and of a polar inclination of 85°.
Aboard the main satellite 12 instruments will be accommodated, which are mainly dedicated for mapping of the lunar surface.
The two sub-satellites with their PRARE-L instrumentation form an experiment for gravity field recovery and carry two additional instruments.
The German scientific institutions and industry is very well positioned for conducting further national and international lunar and planetary missions.

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1. Beitrag des FE-Ergebnisses zu den förderpolitischen Zielen:
Die Ergebnisse der Phase A zeigen das technische Konzept und die Machbarkeit einer nationalen Mondmission, die existierende Spitzentechnologie und -forschung nutzt und fördert. Insbesondere bezüglich der Instrumente zeigt die Studie ein großes Förderpotential der Mission auf, die weltweit anerkannte Expertise der deutschen Raumfahrtindustrie zu erhalten und auszubauen.

2. Wissenschaftlich-technische Ergebnisse des FE-Vorhabens:
Im Rahmen der Studie wurden die wissenschaftlichen Anforderungen an eine, im internationalen Vergleich führende, Mondmission definiert. In enger Kooperation dazu stand die Entwicklung der technischen Konzepte der Nutzlasten und des Missionskonzepts.
In der Studie wurde die wissenschaftliche Bedeutung der Mission sowie deren technische Realisierbarkeit durch die deutsche Industrie aufgezeigt. Es wurden die notwendigen Folgeuntersuchungen und der Entwicklungsbedarf für die Durchführung der Mission identifiziert.

3. Erfindungen/Schutsrechsanmeldungen:
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4. Wirtschaftliche Erfolgsaussichten nach Auftragsende:
Direkte Verwertung der Ergebnisse in künftigen Explorations-Aktivitäten

5. Wissenschaftlich/Technische Erfolgsaussichten nach Auftragsende:
Im Rahmen der Studie wurden die wissenschaftlichen Grundlagen für zukünftige lunare Beobachtungsmissionen definiert, die unmittelbar in internationale Kooperationsprogramme eingebracht werden können.

6. Eventuelle wissenschaftliche und wirtschaftliche Anschlussfähigkeit für eine mögliche notwendige nächste Phase:
Die Phase A sollte hinsichtlich der Instrumententwicklung weiter geführt werden.

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Table of Contents

1 Introduction

1.1 Foreword

This document summarises the results of Lunar Exploration Orbiter (LEO) Phase A study, which have been obtained from April 2008 up to the cancellation in August 2008, followed by a run-down until October/November 2008. The results reflect the analysis carried out in the concept phase of the study (first part of the study) and the amending work that could be carried out within the scope of the run-down of the study. Due to the premature cancellation of the phase A the design phase (second part of the study) could not be conducted. Despite the short duration of the activities several key results have been obtained:

- The mission and system concept of the LEO Phase 0 study, that had been led by Astrium GmbH, was confirmed in all major aspects.
- The instrument concepts have been consolidated und critical items have been identified.
- The main satellite concept has been refined to simplify the interfaces between instruments and platform both during integration and testing on ground and during operations.
- The concept of a subsatellite pair for gravimetry and magnetometry has been consolidated and been verified as the only solution with a scientifically attractive performance.
- The orbit analysis has been refined and the use of stable orbits in order to save propellant while maintaining full science performance has been confirmed.
- The mission performance has been analysed in detail, with focus on the major science goals of the mission.

Despite these successes, the current status of the LEO phase A study leaves several issues unaddressed. Compared to the phase 0 the instrument suite of LEO for the phase A study had been expanded by three additional payload to a total of 15 instruments. The newly selected instruments were:

- LEVIS: A video camera for public outreach
- ATON: An autonomous vision-based navigation sensor and processor
- XRF-L: An X-ray spectrometer

Besides this enlargement of the instrument suite, the concept consolidation of the other instruments has led to a significant increase of the mass and power consumption of the payload. A further mass increase has been induced by the equipment of the payload support system that has been introduced to simplify the payload to platform interfaces. As a result the mass margin of both, the main satellite and the subsatellites, have shrunk to a level that is not acceptable in a phase A. While this does not raise principle questions about mission feasibility it underlines the need for decisions to consolidate to the design of the instrument complement. In particular the following options to consolidate the resources could be considered:

- Strict mass and power apportionment for each instrument: This would allow to maintain the whole instrument suite but would most likely compromise the performance of the individual instruments.
- Reduction of the number of instruments: This would allow maintaining the full performance of the instruments that have passed the down-selection process.

Also several spacecraft subsystems still have potential for resource reduction. This should be pursued during the design phase when the phase A study is resumed. On the other hand some subsystems need further analysis and consolidation that may in effect result in an increased resource demand for these subsystems. Most notably this holds for the thermal control system (of both, main satellite and subsatellites) for which the varying thermal environment on the Moon poses significant challenges.

A change of the lunar transfer orbit concept or baseline launcher to increase the available mass is not considered as the appropriate answer to the current resource increase. The main reason is, that while a change of transfer concept could enhance the available mass, it would not help to overcome power limitations. An enlargement of the solar array would still be prohibited by the strict pointing requirements of the LEO payload which in turn limit the maximum size of the single solar array.

The approach to resources consolidation should also take into account cost aspects. A cost estimate of the elements of the LEO mission was only foreseen for the design phase of the LEO phase A study, which has not been performed. In order to remain within the planned cost envelope of the LEO mission, not all approaches to a consolidation of the space segment resources may be feasible.

This report clearly demonstrates that the LEO mission is basically feasible and all primary science goals could be fulfilled. While already this report contains a considerable level of detail, the reader is referred to the Final Data Package of the LEO Phase A study for the detailed analysis and specification of the elements of the LEO mission and system. In particular it is worth stressing that a comprehensive performance report [RD25] has been established that demonstrates in detail that the LEO mission could meet its demanding scientific goals after having performed a consolidation of the resource issue.

1.2 References

1.2.1 Applicable Documents

[AD1] LEO-MI-ASG-TN-003, Issue 1, Abbreviations

[AD2] LEO-SD-DPF-SP-001, Issue 1, System Functional Specification

[AD3] LEO-SY-ASG-SP-001, Issue 1, System Technical Specification

[AD4] LEO-SY-ASG-SP-002, Issue 1, General Design and Interface Requirements

1.2.2 Reference Documents

[RD1] LEO-RM-UKI-IF-001, Issue 1, RadMo ICD

[RD2] SERTIS Optical Design (ZEMAX File), H. Hirsch, Astro- und Feinwerktechnik Adlershof GmbH / DLR, Berlin

[RD3] SER-DLR-AN-001, Optical Performance Analysis (MERTIS design), Issue 1, June 2008

[RD4] LEO USMI Trade-off Discussion, Technical Note, Institut für Astronomie und Astrophysik der Universität Tübingen (IAAT), USMI-IAAT-TN-001, Issue 2

[RD5] LEO-VN-JOP-DD-001, Issue 1, VIS-NIR Design Description

[RD6] NEXT-LL-LES-ESA(HME)-0001, NEXT Phase A Mission Study, Lunar Environment Specification

[RD7] Flechtner, F., Neumayer, K.-H., Kusche, J., Schäfer, W., Sohl, F., Simulation study for the determination of the lunar gravity field from PRARE-L tracking onboard the German LEO mission, Advances in Space Research 42, pp. 1405-1413, doi: 10.1016/j.asr.2008.06.003, 2008.

[RD8] Foing B.H., MORO – Moon Orbiting Satellite, Phase A Study Report, ESA Doc. SCI(96)1, March 1996.

[RD9] Heiskanen, W.A. and H. Moritz, Physical Geodesy, W.H. Freeman and Co., 1967.

[RD10] Konopliv A.S., Asmar, S.W., & Yuan, D.N., Recent gravity field models as a result of the Lunar Prospector mission, Icarus, 150, 1-18, 2001.

[RD11] Lemoine, F.G.R., Smith, D.E., Zuber, M.T., Neumann, G.A., & Rowlands, D.D., A 70th degree lunar gravity model (GLGM-2) from Clementine and other tracking data, J. Geophys. Res. (Planets), 102 (E7), 16339-16359, 1997.

[RD12] Reigber, C., Balmino, G., Schwintzer, P., Biancale, R., Bode, A., Lemoine, J.M., König, R., Loyer, S., Neumayer, K.H., Marty, J.C., Barthelmes, F., Perosanz, F., Zhu, S.Y., A high-quality global gravity field from CHAMP GPS tracking data and accelerometry (EIGEN-1S), Geophys. Res. Let., 29 (14), doi: 10.1029/2002GL0115064.

[RD13] Reigber, Ch., Schmidt, R., Flechtner, F., König, R., Meyer, U., Neumayer, K. H., Schwintzer, P., & Zhu, S. Y., An Earth gravity field model complete to degree and order 150 from GRACE: EIGEN-GRACE02S, J. Geodyn., 39, 1-10, 2005.

[RD14] Tapley, B.D., & Reigber, Ch., The GRACE Mission: Status and Future Plans, EOS Trans. AGU, 82(47), fall Meeting, Suppl. G14C-02, 2001.

[RD15] Wiezcorek, M.A., The gravity and topography of terrestrial planets, Treatise on Geophysics (Editor-in-Chief G. Schubert), Volume 10, Planets and Moons (Ed. T. Spohn), 165-206, 2007.

[RD16] Zuber, M.T., Smith, D.E; Lemoine, F.G., & Neumann, G.A., The shape and internal structure of the Moon from the Clementine mission, Science, 266 (5192), 1839-1843, 1994.

[RD17] LEO-SY-AST-TN-001, Issue 1.2, Lunar Exploration Orbiter Transfer Analysis

[RD18] LEO-SY-ASG-TN-002, Issue 1, Mission Analysis Report

[RD19] Locoche, S. Eccentricity Vector Control for the German Lunar Exploration Orbiter Mission, Diploma Thesis, 2008

[RD20] LEO-MI-ASG-FR-003DD-004, Issue 1, Mission Operations Concept

[RD21] LEO-PF-OHB-TN-003, Issue 1, Electrical Analysis and Budgets for LEO Phase A

[RD22] Christou, A.A. J. Oberst, D. Koschny, J. Vaubaillon, J. McAuliffe, C. Kolb, H. Lammer, V. Mangano, M. Khodachenko, B. Kazeminejad, and H.O. Rucker, Comparative studies of meteoroid-planet interaction in the inner solar system, Planet. and Space Science Vol. 55, Issue 14, 2049-2062, 2007.

[RD23] Selsis, F., M. Lemmon, J. Vaubaillon, and H.F. Bell, Extraterrestrial meteors: A martian meteor and ist parent comet, Nature 6, 581, 2005

[RD24] LEO-SD-MPS-TN003, Mall, U: Potential other uses of VIS-NIR

[RD25] LEO-SY-ASG-DD-003, Issue 1, System Performance Report

2 Concept Overview

2.1 Top-level science goals

Introduction

The German initiative for the Lunar Exploration Orbiter (LEO) originated from the national conference "Exploration of our Solar System", held in Dresden in November 2006. Major result of this conference was that the Moon is of high interest for the scientific community for various reasons: it is affordable to perform an orbiting mission to the Moon and it insures technological and scientific progress necessary to assist further exploration activities of our Solar System. Based on scientific proposals elaborated by more than 50 German scientists in January 2007, payload of 13 scientific instruments, one technology demonstration payload and one auxiliary payload was defined and selected. Further, a mission assessment study performed by the German industries demonstrated the feasibility of a national lunar mission.

LEO is planned to be launched in 2012 and shall orbit the Moon for about four years at low altitude (~50 km) in order to map the Moon geomorphologically, geochemically and geophysically with resolutions down to 1m globally.

Figure 2.1-1 LEO Main Satellite with Sub-satellites

Overall scientific approach and justification

The Moon has been visited and studied in the past, we even have samples from the Moon, but there are unanswered questions concerning

- the Moon's origin: is it compelling essential or random?

- the internal structure: how does the lunar mantle zone work and does the Moon have a core?
- the reason for the dichotomy: why is the farside crust thicker; why are the centre of mass and the centre of figure apart; what is the composition of the highland crust and how are rock forming minerals distributed in the highland crust?
- the volcanic evolution: how does the composition of lava change with time; how does dense magma work through a less dense crust; what is the time scale of volcanism, what are the sources of volcanism?
- the origin of mass concentrations (mascons)?
- the origin of remnant magnetization?
- the characteristics of the lunar debris cover (regolith) and its transition to underlaying bedrock?
- interaction with the interplanetary environment: what is the characteristics of the lunar exosphere; how does the atmosphereless Moon react with radiation and interplanetary particles?

The Moon is an integral part of the Earth-Moon system, it is a witness to more than 4.5 billion years of solar system history, and it is the only planetary body except Earth for which we have samples from known locations. The vast amount of knowledge gained from the Apollo and other lunar missions of the late 1960's and early 1970's demonstrates how valuable the Moon is for the understanding of our planetary system. Even today, the Moon remains a scientifically and technologically extremely interesting target as many open questions about the Earth-Moon system still unanswered, even though new data since Apollo have addressed some of them. Therefore, returning to the Moon is the critical stepping-stone to further exploring our immediate planetary neighbourhood.

Understanding the origin and evolution of the terrestrial planets including Earth requires information about their early differentiation, volcanism and related tectonic processes. However, the physics and chemistry of these processes and its chronological sequences are not completely known and can not be deciphered on the dynamic Earth, the young Venus and the weathered Mars. The Moon's composition is, due to the lack of water and its restricted geological active phase, relatively simple and thus provides insight into planetary processes that are much more obscured on other bodies. In particular, Earth and Venus exhibit extremely young surfaces, containing almost no record of the early evolution of a planet. Thus, evidence on how planets differentiate, of how early magma oceans operate as well as on secondary differentiation and initial volcanism is restricted to the Moon. Thus, planetary processes can be studied on the Moon in its original state, making the Moon the simple endmember of planetary evolution.

Earth and Moon form a common tidally evolved planetary system that is unique among the terrestrial planets. Is there a direct correlation of the specific evolution of Earth including life and the existence of the Moon? The Moon is thought to be the product of an early planetary collision of a Mars-sized body with Earth. However this model needs to be confirmed by measurable “truth”. Key for this truth is the knowledge of the Moon's composition and its global distribution.

Dating of planetary surface and thus of planetary processes like emplacement of lava, collision events, and breaking of the crust depends on the distribution and frequency of impact craters. This statistical method is based on the long record of impacts known from the lunar surface and correlations with the absolute age of lunar samples. However, particularly small impact craters that are needed to improve

the accuracy of this dating method are not mapped out globally on the Moon. As the Moon has no atmosphere its surface will not only collect impacts of smallest scale but is hit by sizes down to the particles of the solar wind. The surface debris called regolith has thus collected information about activities in our space environment over time until the beginning.
In order to address the open questions of the Earth-Moon system a necessary further step in investigating the Moon is getting a global and integrated view of its geology, geochemistry and geophysics at highest resolution down to meter scale. In particular, we need to significantly improve our understanding of the lunar surface structure and composition, surface ages, mineralogy, physical properties, interior, thermal history, gravity field, regolith structure, and magnetic field. A low altitude orbiting spacecraft, equipped with a wealth of high-resolution remote sensing instrumentation, can achieve such a goal. Highest resolution geological, geochemical and geophysical mapping will provide the unambiguously needed information to plan landings and future utilization of the Moon.

International context

Numerous space-faring nations have realized and identified the unique opportunities related to lunar exploration and have planned missions to the Moon within the next few years. Among these missions, LEO will be unique, because it will globally explore the Moon in unprecedented spatial and spectral resolution: 1.25 m in stereo, 12 m spectrally in the UV and 24 m in the NIR (0.2 – 3 µm), 200 m in the thermal infrared (7 – 14 µm), subsurface sounding at 6 m resolution, and 20 km resolution for lunar gravity with an accuracy of < 0.1 mGal. Therefore, LEO will significantly improve our understanding of the lunar surface composition, surface ages, mineralogy, physical properties, interior, thermal history, gravity field, regolith structure, and magnetic field. The Lunar Exploration Orbiter will carry an entire suite of innovative and complementary technologies, including a high-resolution stereo camera system, several spectrometers that cover previously unexplored parts of the electromagnetic spectrum over a broad range of wavelengths, microwave and radar experiments, a sensitive magnetometer and, and two sub-satellites for gravitational studies. The Lunar Explorations Orbiter concept is technologically challenging and will gather unique, integrated, interdisciplinary data sets that are of high scientific interest and will provide an unprecedented new context for other international lunar missions. With its high visibility, LEO will foster the growing acceptance of space exploration in Germany and will capture the imagination of the general public.

Unique features

The most visible mission goal of LEO will be the global mapping of the lunar surface with high spatial as well as spectral resolution. Therefore, in addition to a stereoscopic global mapping in the meter range, a screening of the electromagnetic spectrum within a very broad range will be performed. In particular, spectral mapping in the ultraviolet and mid-infrared will provide insight into mineralogical and thermal properties so far unexplored in these wavelength ranges. Fine scale analysis of the lunar regolith by radar sounding will provide structural information about regolith layering. The determination of the dust distribution in the lunar orbit will provide information about processes between the lunar surface and exosphere supported by direct observations of lunar flashes. The geophysical properties of the Moon will be investigated by recording the magnetic and gravitational field with so far unrivalled accuracy due to the low orbit, stable sub-satellites and specific tracking. Measuring of the radiation environment will finally complete the exosphere investigations. Combined observations based on simultaneous instrument adjustment and correlated data processing will provide an integrated geological, geochemical and geophysical database that enable

- the exploration and utilization of the Moon in the 21st century;
- the solution fundamental problems of planetology concerning the origin and evolution of terrestrial bodies;
- understanding the uniqueness of the Earth-Moon System.
- the absolute calibration of the impact chronology for the datation of solar system processes.
- deciphering the lunar regolith as record for space environmental conditions.
- mapping lunar resources.

LEO is featuring a set of unique scientific capabilities w.r.t. other planned missions including: (1) 100% global coverage of he lunar surface with all remote sensing instruments with stereo resolutions of close to 1 m and spatial resolution of the spectral bands of < 200 m in the MIR down to < 12 m in the UV. (2) Besides the VIS-NIR spectral range so far uncovered wavelengths in the ultraviolet (0.2 – 0.4 µm) and mid-infrared (7 - 14 µm) will be globally mapped. (3) Global coverage and subsurface detection of the regolith with vertical resolutions of about 3 m down to a few tens of meters (high resolution Synthetic Aperture Radar with 25cm wavelength) and on mm-scale within the first 2 m (microwave-instrument) will investigate the regolith's structure. (4) Detailed measurements of the global lunar gravity field < 0.1 mGal accuracy at 20 km spatial resolution and magnetic field with 0.1 nT accuracy from a low orbit (~50 km) using two sub-satellites and simultaneous Earth tracking, supported by a radiation monitor and two independent magnetometers, will provide high precision and, in addition, will enable to geophysically investigate the lunar far side. (5) The long mission duration of 4 years yields multiple high resolution stereo coverage and thus monitoring of new impacts; this is supported by a flash detection camera searching directly for impact events and dust detection in the exosphere.
In summary, the overall mission scientific objectives and major measurement requirements of LEO are as follows:

Mission Scientific Objectives

- Provide an integrated geoscientific global data base for the future exploration and utilization of the Moon
- Solve fundamental problems of planetology
  - The origin and geoscientific evolution of the Moon as baseline for the understanding of the terrestrial planets.
  - Uniqueness of the Earth-Moon System.
  - Surface-space interaction and space environment.
  - Absolute calibration of the impact chronology for the datation of solar system processes.
  - Regolith as record for space environmental conditions.
- Map lunar Resources in a cartographical, geological, mineralogical, geophysical sense.
- Provide a high-resolution road map for further exploration.

Major Measurement Requirements

- Global coverage in all wavelength ranges
- Spectral coverage: *X-Ray*, *UV, VIS, NIR, MIR, Microwave, Radar*
- Highest spatial resolution with ground sampling distances (at 50 km orbit altitude, c.p. Fig. 2.1-2):

1.25 m stereo,
12 m (UV),
24 m (VIS, NIR),
< 200 m (MIR, < 1 km thermal),
~ 6 m (Radar),
~ 6 km (Microwave, X-Ray)

- Sub surface sounding
  - few meters deep with mm resolution (Microwave)
  - up to hundred meter deep with m resolution (Radar)
- Physical characterization of the regolith (composition, physical properties, thermal, polarization, scattering, maturity, space weather interaction, particle properties)
- Global gravity with 0.1 mGal accuracy at 20 km spatial resolution
- Magnetic field globally with 0.1 nT accuracy at 20-80 km altitude
- Characterization and monitoring of the lunar space environment regarding radiation, dust and magnetic field

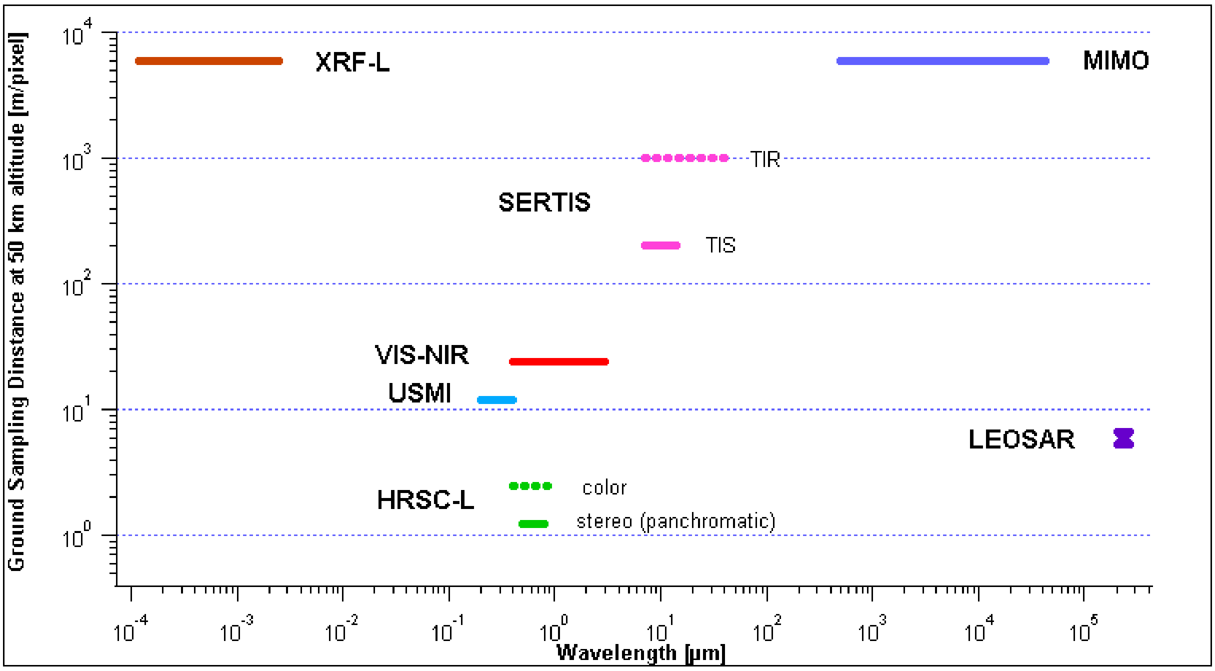


Figure 2.1-2 LEO lunar surface mapping as a function of wavelength coverage and ground sampling distances at an altitude of 50 km.

An overview of the complementary lunar surface mapping instruments onboard LEO as a function of wavelength and ground sampling distance is given in Fig. 2.1-2. The different spectrometer instruments cover a broad range of wavelengths from X-ray to the thermal infrared in order to decipher the composition or the lunar surface with respect to its elemental as well as its mineralogical composition. In each wavelength range of the electromagnetic spectrum, distinct physical processes are responsible for the scattering, absorption and/or emission of radiation and the measurements are highly complementary (cp. Fig. 2.1-3):

1. X-Ray Fluorescence measures the emission of X-rays due to the excitation by solar X-ray radiation and is diagnostic for the abundancies of geochemical elements like Na, Mg, Al, Si, Ca, Ti, Mn, Fe.
2. In the UV, spectral features are caused mainly by charge transfers between (i) oxygen and metal atoms and (ii) metal to metal atoms
3. In the VIS to NIR, spectra are dominated by crystal field transitions within distorted d-shells of the transition metals and especially diagnostic for the mafic mineral groups of olivine and pyroxene.
4. In the thermal infrared, spectral features are related to vibration modes of the silica-oxygen bonds and diagnostic for the silica-related mineralogy

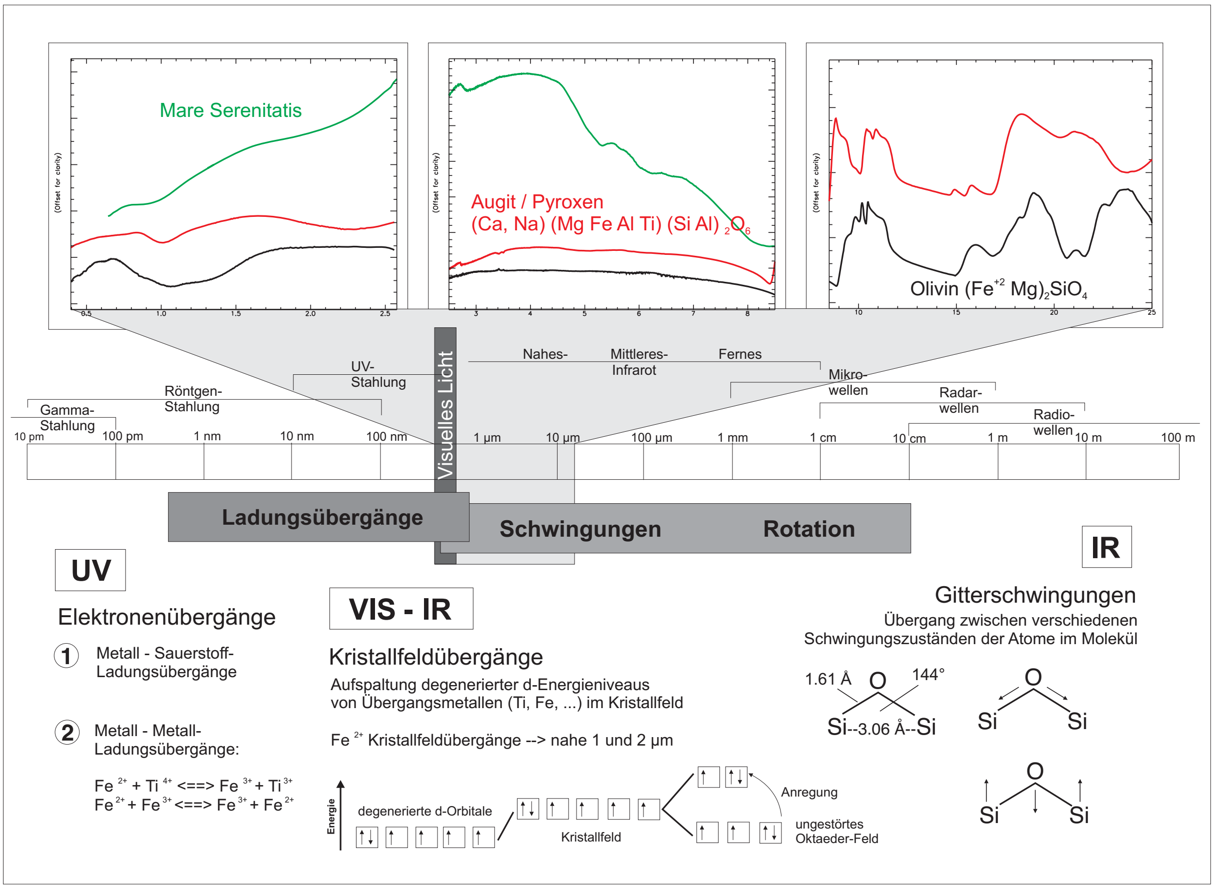


Figure 2.1-3 Physical processes responsible for absorption, reflection and/or emission of electromagnetic radiation in the UV to thermal infrared and the respective spectral signature of minerals and the lunar surface.

2.2 Baseline Concept

The baseline concept for LEO has already been established in the phase 0 study by Astrium GmbH and has been confirmed in all ist major aspects during the concept pahse of the phase A study. This section recalls the basic considerations that have led to the LEO baseline concept in order maintain the rationale for the design fully traceable.

The optimal concept for LEO has to fulfil both the requirements, of the multispectral imaging instruments and that of the instruments for the study of the physical properties of the Moon. This is a particular challenge because the two instrument classes have very distinct and in parts contradicting requirements:

- The multispectral instruments serve the global mapping of the lunar surface and thus have to be pointed towards it. Leaving aside the microwave instruments, LEOSAR and MIMO, which have an off-nadir pointing direction, the pointing of all mapping instruments is towards nadir. Amongst the optical Instruments the HRSC-L is the driver for the requirements on the absolute measurement error and in particular for the relative pointing error due to its high resolution and large dynamical range. Also the other multispectral instruments require a satellite platform with precise attitude control.
- The main requirement of the magnetic field measurement and the gravimetry on the satellite platform are an unperturbed measurement environment. For magnetometry this means a high magnetic cleanliness of the platform and in particular a minimisation of unknown and variable magnetic fields originating from the satellite. This is most efficiently implemented by reducing the number of potentially disturbing sources. Gravimetry via satellite-to-satellite tracking relies on a precise determination of the velocity between the centres of mass of the two satellites. The measurement is however conducted between the phase centres of two antennas which in general cannot be place into the centre of mass of the satellite. Hence a precise knowledge of the velocity between the centre of mass of the satellite and the antenna are needed. The major contribution to the uncertainty in this velocity stems from the motion of the centre of mass of the satellite due to fuel sloshing and moving parts such as a pointable solar array. In consequence a satellite platform suitable for gravimetry should avoid moving parts and carry minimal fuel.

During the phase 0 a concept for LEO has been established that fulfils all major requirements of the instruments and while at the same time it minimises system complexity.

The concept foresees a system of one Main Satellite and two Subsatellites. The Main Satellite carries all scientific instruments with the exception of the magnetometry, i.e. LunarMag, the gravimetry, i.e. PRARE-L and the gravimetry support instrument RaPS.

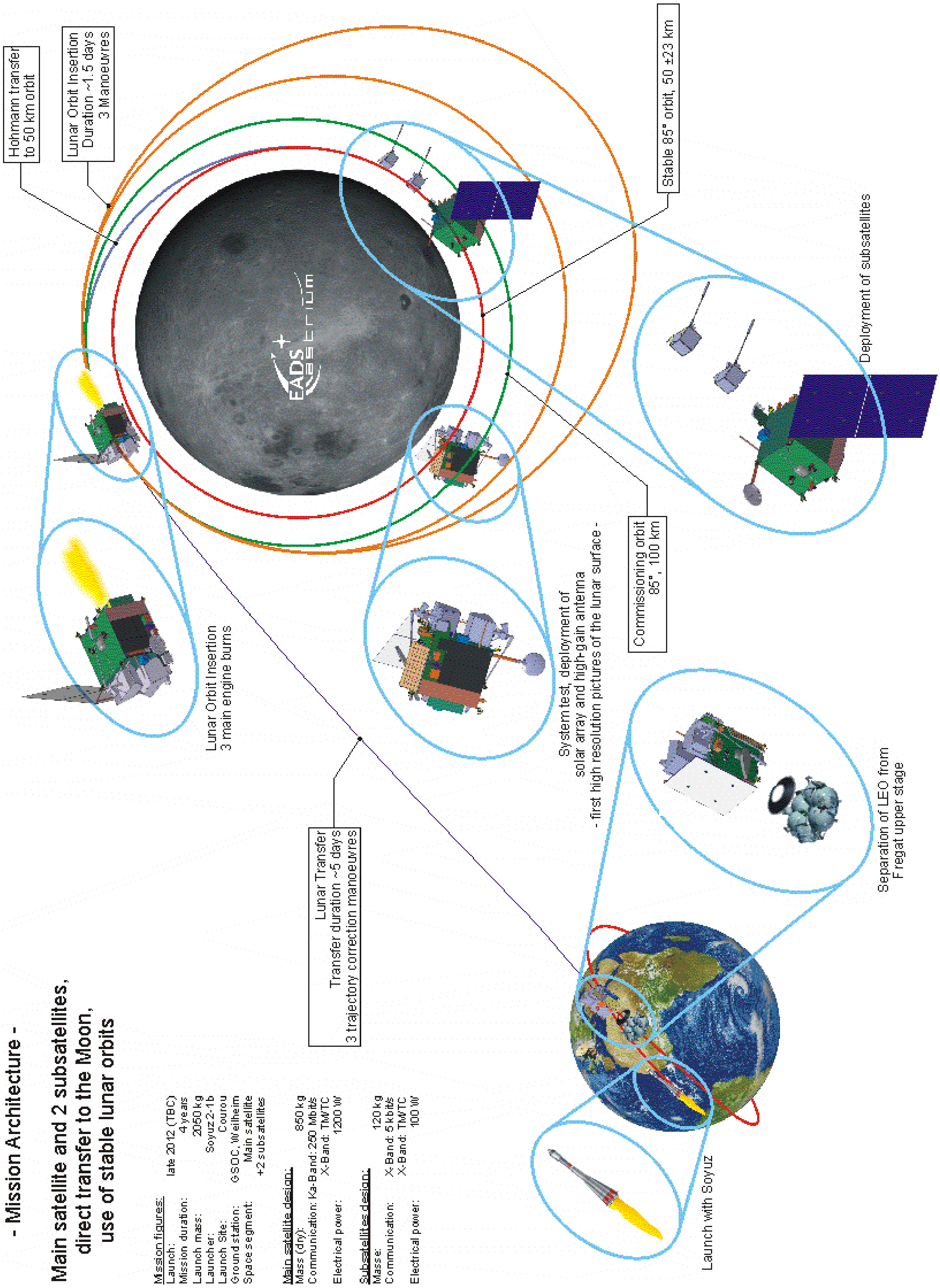
Another unique feature of the LEO mission concept compared to other lunar missions is the choice of operational orbit. The LEO main satellite will fly in a lunar orbit of 50 km altitude and 85° inclination for 3 years and then carry out an inclination change into a polar orbit of 50 km altitude. In contrast to the polar orbit, the 85° orbit is stable and hence drastically reduces the propellant demand for orbit maintenance during the first 3 years of the mission. The subsatellites remains in the stable 85° orbit throughout their 4-year mission.

Mission Concept

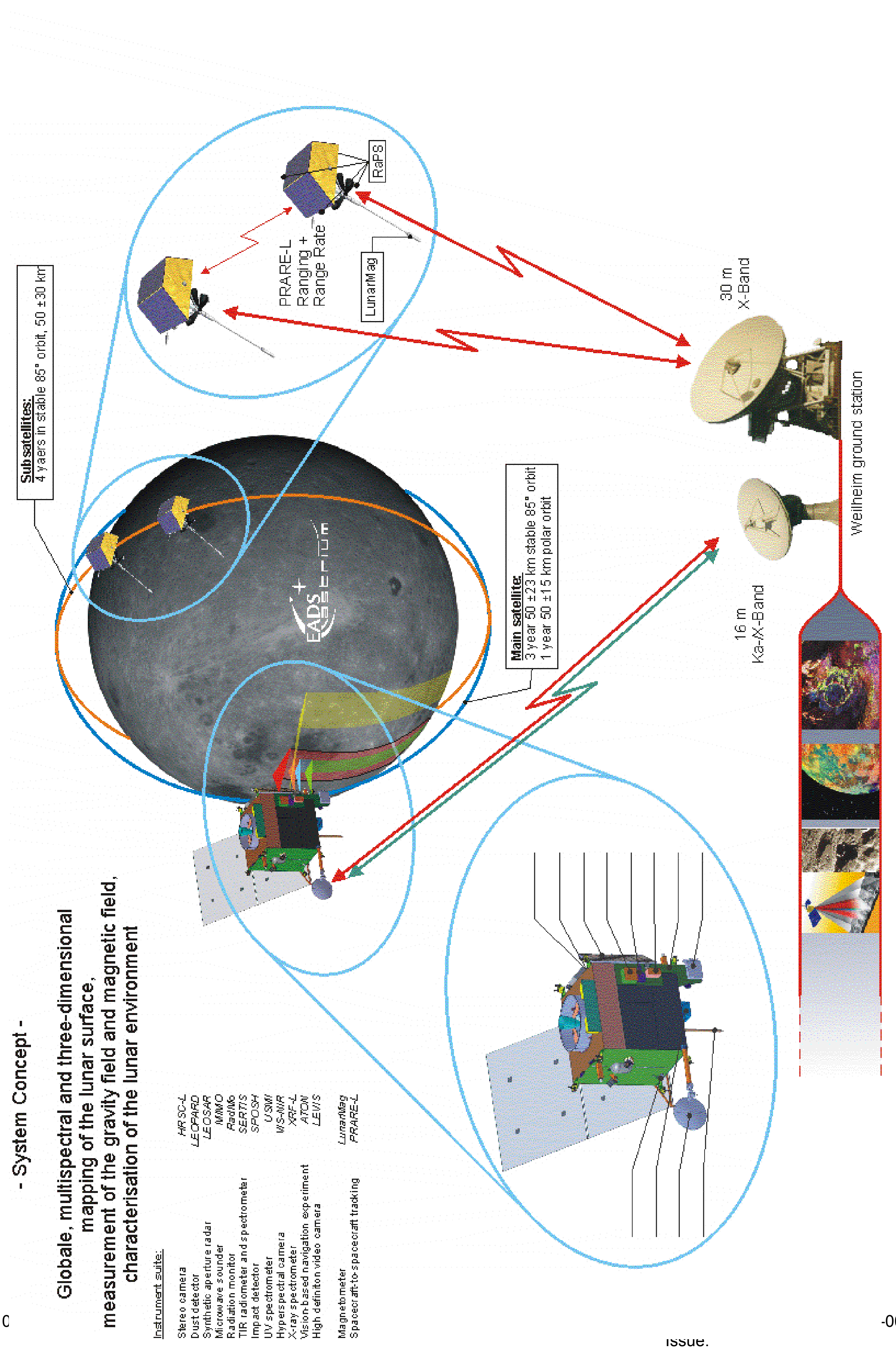
2.2.1 Mission

The baseline concept of LEO is displayed in graphical form on the two following pages. The space segment consists out of one main satellite and two subsatellites. The three satellites are launched as a stack on a single Soyuz launcher. Soyuz was selected as the baseline launcher because it is the smallest launcher that is suitable for the LEO mission and because it has a unparalleled reliability. With a Soyuz and a Fregat upper stage a direct injection into a lunar transfer orbit is possible. The direct transfer is the solution with the lowest operational complexity and is hence preferred over a transfer via a highly ellipitical orbit that would yield a slightly higher mass in lunar orbit.

The lunar orbit insertion manoeuvre is carried out by the main satellite. The commissioning orbit is reached by a total of three burns of the 400 N bi-propellant main engine. The commissioning orbit has an altitude of 100 km and an inclination of 85°. After a commissioning phase of approximately two months the orbit is lowered to the first science orbit at an altitude of 50 km, also having an inclination of 85°. Both the commissioning and the first science orbit are chosen to be long-term stable such that the LEO can remain in them for a practically unlimited period without manoeuvres. An operational strategy for LEO has been worked out that makes sure that the stable orbits are actually attainable. The subsatellites are deployed in the 85° orbit. After roughly 3 years the main satellite carries out an inclination change into a polar orbit. The polar orbit is maintained for at least one year until the end of the nominal mission. The two subsatellites remain in the 85° orbit. For the extension of the main satellite mission a stable elliptical polar orbit or a return into the 85° orbit could be considered.



- Mission Architecture -
Main satellite and 2 subsatellites, direct transfer to the Moon, use of stable lunar orbits
Mission figures:
Launch: late 2012 (TBC)
Mission duration: 4 years
Launch mass: 2050 kg
Launcher: Soyuz 2-1b
Launch Site: Courou
Ground station: GSOC, Weilheim
Space segment: Main satellite +2 subsatellites
Main satellite design:
Mass (dry): 850 kg
Communication: Ka-Band: 250 Mbit/s X-Band: TM/TC
Electrical power: 1200 W
Subsatellites design:
Masse: 120 kg
Communication: X-Band: 5 kbit/s X-Band: TM/TC
Electrical power: 100 W
Launch with Soyuz
Separation of LEO from Fregat upper stage
Lunar Transfer
Transfer duration ~5 days
3 trajectory correction manoeuvres
Lunar Orbit Insertion
3 main engine burns
System test, deployment of solar array and high-gain antenna
- first high resolution pictures of the lunar surface -
Commissioning orbit
85°, 100 km
Stable 85° orbit, 50 ±23 km
Deployment of subsatellites
Hohmann transfer to 50 km orbit
Lunar Orbit Insertion
Duration ~1.5 days
3 Manoeuvres
EADS astrium


- System Concept -
Globale, multispectral and three-dimensional mapping of the lunar surface, measurement of the gravity field and magnetic field, characterisation of the lunar environment
Instrument suite:
Stereo camera HRSC-L
Dust detector LEOPARD
Synthetic aperture radar LEOSAR
Microwave sounder MIMO
Radiation monitor RadMo
TIR radiometer and spectrometer SERTIS
Impact detector SPOSH
UV spectrometer USMI
Hyperspectral camera WS-NIR
X-ray spectrometer XRF-L
Vision-based navigation experiment ATON
High definiton video camera LEVIS
Magnetometer LunarMag
Spacecraft-to-spacecraft tracking PRARE-L
Subsatellites:
4 yaers in stable 85° orbit, 50 ±30 km
Main satellite:
3 year 50 ±23 km stable 85° orbit
1 year 50 ±15 km polar orbit
PRARE-L
Ranging +
Range Rate
LunarMag
RaPS
16 m
Ka-/X-Band
30 m
X-Band
Weilheim ground station
EADS astrium

The timelining of the instrument operations is a critical aspect in the mission planning. The ON-times of the instruments are primarily limited by the data rate that is available for science data downlink. The LEO system was hence tuned to find the optimal combination of mission timeline and telecommunication architecture. For the design the following aspect have been taken into consideration:

- The top-level requirement of global coverage
- A limitation of the overall mission duration to 4 years
- The observation requirements of the instruments (e.g. illumination conditions)
- The available telecommunications bandwidth in various frequency ranges (in particular the ITU regulations have been taken into account)
- The available telecommunications hardware

The key result is, that global coverage can be accomplished within four year using one Ka-band ground station in Germany, if the instruments are group into instrument sets that are operated in alternating fashion.

2.2.2 Space Segment

The LEO baseline system consists out of 3 satellites, one main satellite and two identical subsatellites. The main satellite has a dry mass of approximately 990 kg and carries the optical instruments, the microwave instruments, the radiation monitor and the dust detector (including the Langmuir probe). The satellite is oriented towards nadir. Its principle features are:

- A bi-propellant propulsion system for lunar orbit insertion and the orbit and attitude control
- A single wing solar array with on degree of freedom
- A two degrees of freedom high-gain antenna
- A highly stable attitude control, in order to fulfil the relative pointing error requirements of the HRSC-L and other optical payloads

The two subsatellites have a mass of approximately 120 kg, each. Each of them carries one magnetometer, a PRARE-L instrument and six radiation pressure sensors (RaPS). They follow an attitude law that coarsely orients the antennas for satellite-to-satellite tracking towards each other. The main features of the subsatellites are:

- Highly intergrate mini-satellite electrical architecture
- A configuration with a trapezoidal cross-section analogous to that of the GRACE satellites
- A deployable boom for the magnetometer, which at the same time supports gravity-gradient stabilisation of the subsatellites
- A cold-gas propulsion system for attitude and orbit control. The use of a cold gas system enables a high stability of the subsatellite centre of mass.
- A combined telecommunications and ranging system using a spread-spectrum signal on the basis of the PRARE/ACES technology.

The main advantages of the baseline concept are:

- The optimal compliance with all scientific requirements and instrument requirements
- High cost efficiency of the main satellite by focussing its design on the requirements of the optical and microwave instruments
- High cost efficiency of the subsatellites by application of a mini-satellite design philosophy

2.2.3 The Payload Suite

The elements of the payload suite are listed in Table 2.2-1 together with their acronyms and their Principle Investigators. The scientific objectives of the instruments have been described in chapter 2.1, above, and the requirements of the experiments is given in chapter 3.2 and 3.3.

Table 2.2-1 Payload Suites of the LEO Main Satellite and of the Sub-satellites

Payload	Acronym	Principle Investigator	Organisation
Instruments on Main Satellite			
High Resolution Stereo Camera - Lunar	HRSC-L	Prof. Dr. R. Jaumann	DLR, Institut für Planetenforschung (PF)
Lunar Exploration Orbiter Dust Particle Detector	LEOPARD	Dr. S. Kempf	Technische Universität Braunschweig, Institut für Geophysik und extraterrestrische Physik
SAR-SOUNDER Instrument	LEOSAR	Prof. Dr.-Ing. A. Moreira	DLR, Institut für Hochfrequenztechnik und Radarsysteme (HR)
Microwave Instrument for a Moon Orbiter	MIMO	Dr. P. Hartogh	MPS, Katlenburg-Lindau
Radiation Monitor	RadMo	Prof. Dr. R. F. Wimmer-Schweingruber	Christian-Albrechts-Universität zu Kiel, Institut für Experimentelle und Angewandte Physik
Selenologic Radiometer and Thermal Infrared Spectrometer	SERTIS	Prof. Dr. H. Hiesinger Dr. J. Helbert	Westfälische Wilhelms-Universität Münster, Institut für Planetologie DLR, Institut für Planetenforschung (PF)
Smart Panoramic Optical Sensor Head	SPOSH-L	Prof. Dr. J. Oberst	Technische Universität Berlin, Institut für Geodäsie und Geoinformationstechnik
Ultraviolet Spectral Mapping Instrument	USMI	Prof. Dr. K. Werner	Universität Tübingen, Institut für Astronomie und Astrophysik
VIS-NIR Mapping Spectrometer	VIS-NIR	Dr. U. Mall	MPS, Katlenburg-Lindau
Lunar X-ray Fluorescence Experiment	XRF-L	Prof. Dr. G. Neukum Dr. S. van Gasselt	Freie Universität Berlin, Institut für Geologische Wissenschaften
Technology Demonstration Payload on Main Satellite			
Autonomous Terrain based Optical Navigation	ATON	Dr. S. Theil	DLR, Institut für Raumfahrtsysteme (RY)
Auxiliary Payload on Main Satellite			
Lunar Exploration Video Imager System	LEVIS	Prof. H. Michalik	Technische Universität Braunschweig, Institut für Datentechnik und Kommunikationsnetze
Instruments on Sub-satellites			
Magnetometer	LunarMag	Prof. Dr. K.-H. Glaßmeier Dr. U. Auster	Technische Universität Braunschweig, Institut für Geophysik und Extraterrestrische Physik
Precise Range and Range Rate Equipment - Lunar	PRARE-L	Dr. F. Flechtner	GeoForschungsZentrum Potsdam, Dep. 1: Geodäsie und Fernerkundung
Radiation Pressure Sensor	RaPS	Dr. K. H. Neumayer	GeoForschungsZentrum Potsdam, Dep. 1: Geodäsie und Fernerkundung

2.2.3.1 The Launch Segment

LEO will be launched on a Soyuz 2-1b with Fregat-M upper stage. The Soyuz/Fregat will be launched from CSG. Prior to launch the LEO space segment will be processed at CSG that includes both fuelling and control centre functions. The launch services are provided by Arianespace. The Soyuz/Fregat will target the lunar transfer orbit insertion point, which puts LEO on a direct trajectory to the Moon. Figure 2.2-1 depicts the LEO Space Segment in the Soyuz small fairing.

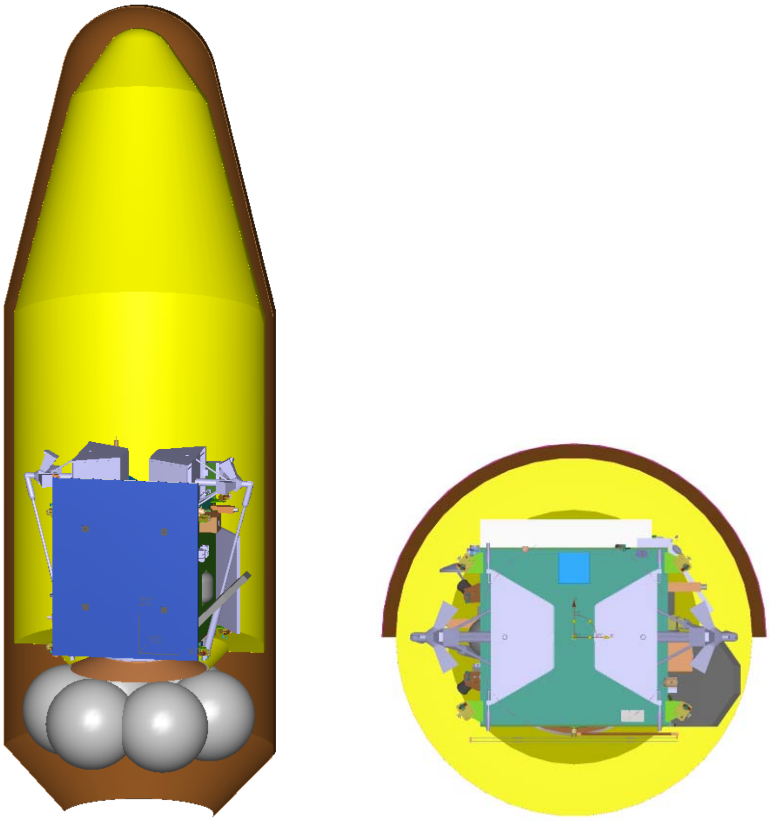


Figure 2.2-1 The launch configuration of LEO Space Segment in the Soyuz fairing

2.2.3.2 The Ground Segment

The LEO ground system includes the Mission Operations System (MOS) and the Science Data System. The ground segment is described in chapter 4.6

Mission Operations System

The LEO Mission Operations System breaks down into five separate elements that include:

- the Mission Planning System (MPS)
- the Flight Operations System (FOS)
- the Flight Dynamics System (FDS)
- the Ground Data System (GDS)
- the Payload Operations System (POS)

Science Data System

The Science Data System comprises the 13 different Science Operations Centres and the Payload Ground Segment. The 13 Science Operations Centres and the operations centre for ATON and LEVIS will be responsible for monitoring their respective instruments, data processing, generating inputs in mission planning and maintaining the instruments. The Payload Ground System will permanently archive the mission raw data, the mission data products as well as mission information such as orbit and attitude information.

3 Payload

3.1 Payload Overview

The LEO mission comprises 15 payloads, from which 12 are accommodated on the main satellite and 3 on the two identical subsatellites (cf. Table 2.2-1). The following gives an overview of the instruments with the objectives, key facts and budgets (w/o margins). The instrument requirements are presented in more detail in Chapter 3.2 and 3.3 below.

Table 3.1-1: HRSC-L Overview

HRSC-L (High Resolution Stereo Camera - Lunar)	
Measurement objective:	Global mapping of the lunar surface for – Ortho image map generation with ~1.25 m GSD – Topographic (stereo) map generation – Photometric characterisation with four colours with ~2.5 m GSD – Polarimetric characterisation with horizontal and vertical polarisation with ~2.5 m GSD
Key requirements:	FR-OBS-8.2.1-040: Global coverage for the standard products FR-OBS-8.2.1-050 & 060: Spectral channels PAN, Blue, Green, NIR1, NIR2 FR-OBS-8.2.1-070: GSD of 1.25 m and 2.5 m FR-OBS-8.2.1-090: Stereo angles between -18.9° and +18.9° FR-OBS-8.2.1-120: Dynamic range
Main design drivers:	5 stereo angles between -18.9° and +18.9° GSD of 1.25 m and 2.5 m Dynamic range from 2° solar incidence on dark material to 90° solar incidence on bright material plus margin within one acquisition SNR of 150, resp. 100 for blue, at 10° solar incidence on dark material Global coverage
Budgets:	Mass: 26 kg + 8 kg radiators Envelope: CH 650 x 360 x 270 mm³; DPA 150 x 250 x 250 mm³ Power: 209 W nominal; 77 W standby Data rate: 3900 Mbit/s raw; 250 Mbit/s compressed
Overview:	Camera Head CH (Baffle, Optics; Structure; FPA), Digital Processing Assembly DPA

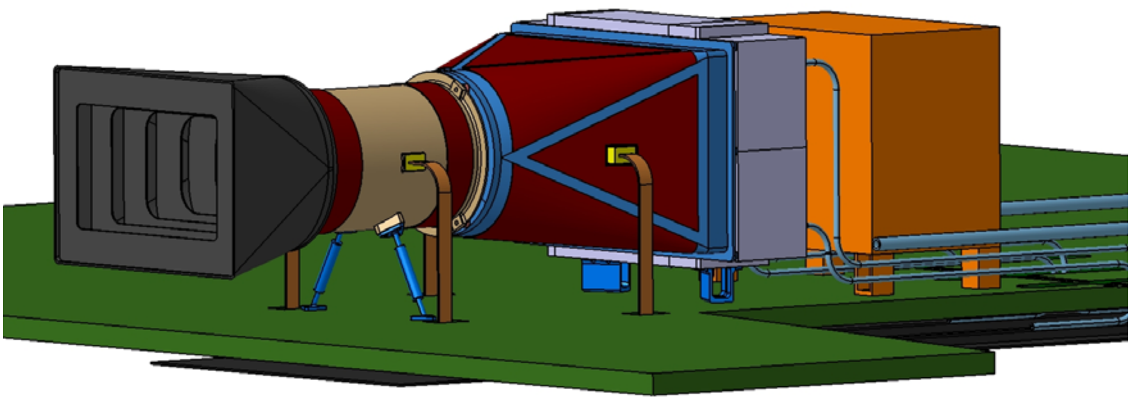


Table 3.1-2: VISNIR Overview

VISNIR (Visible and Near Infrared Imaging Spectrometer)	
Measurement objective:	Global mapping of the lunar surface for – Lunar mineralogy – Investigation on spectral footprints of different lunar minerals – Correlation between lunar morphology and mineralogy
Key requirements:	FR-OBS-8.2.13-010: Global coverage, App. 12m SSD@Nadir FR-OBS-8.2.13-050: Pushbroom scanning FR-OBS-8.2.13-060: Spectral range 400 - 3000 nm FR-OBS-8.2.13-070: Spectral sampling 10 nm FR-OBS-8.2.13-090: Field of view +/- 3.44° (swath width 6 km) FR-OBS-8.2.13-100: 24m GSD FR-OBS-8.2.13-120: Spectral sampling width < 1x spectral sampling interval FR-OBS-8.2.13-130: Spectral sampling width accuracy < 10% FR-OBS-8.2.13-180: SNR > 150 at polar radiance, >400 at equatorial
Main design drivers:	Spectral range (3 µm) not supported by single detector. Spectral sampling interval reduces throughput. Spectral sampling width accuracy < 10% places harsh requirements on thermal and mechanical stability. SNR requirements are difficult to reach with current GSD and spectral sampling
Budgets:	Mass: 19,5 kg + TBD kg radiators Envelope: CH 350 x 300 x 200 mm³; E-Box 250 x 200 x 200 mm³ Power: 40 W nominal; TBD W standby Data rate: Mbit/s raw; 64,5 Mbit/s compressed
Overview:	– TMA based telescope – Offner spectrometer based on a convex grating – Off-the shelf backthinned MCT detector, 500*256 elements (320*256, TBC), 2.5µm cut-off TEC cooled FPA (fallback passively)

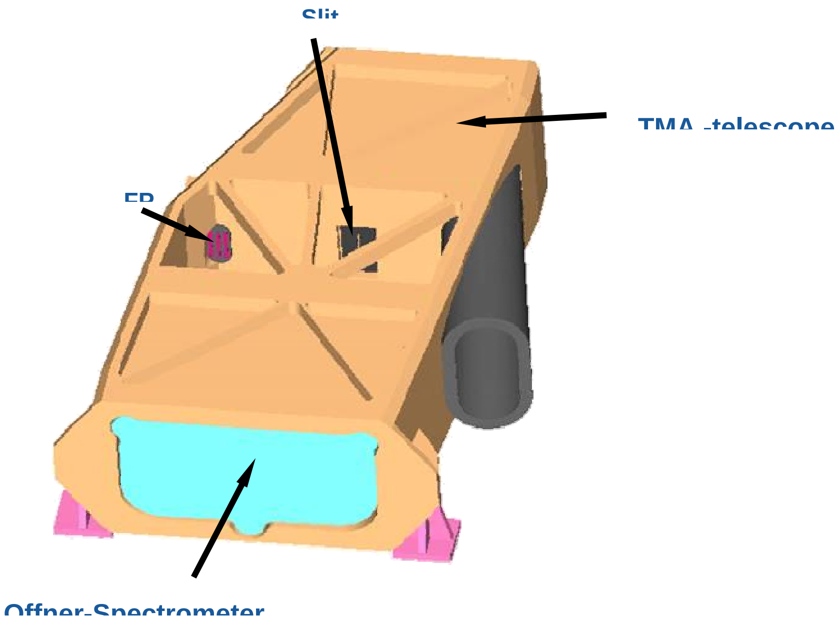


Table 3.1-3: LEOSAR Overview

LEOSAR (Lunar Exploration Orbiter Synthetic Aperture RADAR)	
Measurement objective:	Global mapping of the lunar surface by a Synthetic Aperture Radar (SAR) in stripmap mode – L-band allows deep penetration into the lunar Regolith up to 20 m – High spatial resolution to facilitate the characterization of fine details of the Regolith – Generation of a digital elevation model (DEM) of the scattering zone in the regolith by repeat pass interferometric measurements – Classification of layers close to surface by polarimetric measurements, possible discovering of ice
Key requirements:	Global Coverage Swath width of 12 km Incidence angle of 35° +/- 5° L-Band Spatial resolution of 6 m in single look NESZ < -25 dB Hybrid polarisation
Main design drivers:	Mass constraints
Budgets:	Mass: 20.9 kg + TBD kg radiators Envelope: Antenna 1500 x 800 x 100 mm³; DEU 400 x250 x 250 mm³; PCU 100 x 160 x 50; USO 84 x 54 x 42; 2 HPA 165 x 320 x 25; 2 RFFE 100 x 50 x20mm³ Power: 80 W nominal; TBD W standby Data rate: 396 Mbit/s
Overview:	

Table 3.1-4: LEOPARD Overview

LEOPARD (Lunar Exploration Orbiter Particle Detector)	
Measurement objective:	Characterize the moons dust environment, by measuring: – Mass, speed vector, electrostatic charge, and chemical composition – Reconstructed grain trajectories and their starting points on the Lunar surface – Density maps of the Lunar dust envelope – Distribution of the dust production on the Lunar surface
Key requirements:	Global characterisation of the lunar dust cloud requires continuous measurements during the mission – acceptable: occasional measurement gaps of <1 day – avoid: repetitive observation gaps >1 month Avoid exposure to Sun light within FOV longer than 3 min (TBC) Protection from conducted and radiated EMI; amplitudes between 1 kHz – 100 kHz (TBC) sensed by the trajectory sensor must be smaller than 0.1 fC (TBC) Protection from mechanical vibrations between 1 kHz – 100 kHz that cause charge amplitudes on the wire sensors >0.1 fC Measurement of the dust speed vector with accuracy better than 0.1% for grains speeds of 2 km/s. Determination of dust mass and charge for grains > 0.1 µm TOF mass spectra with resolution better than 100 in a mass range between 1 and 200 u FOV of ±60 with respect to the instrument boresight Langmuir probe: – measurement of the electrostatic potential accuracy: min 0.1V 1 min time resolution at the spacecraft location – measure the local number density and temperature of the plasma electrons in the vicinity of the Moon with accuracy better than 10% – to be mounted on a boom of about 1m length
Main design drivers:	Sun avoidance; EMI requirements; Measurement gap avoidance currently not feasible due to yaw flip. Second instrument can not be accommodated.
Budgets:	Mass: 6.464 kg + TBD kg radiators; Envelope: 316.1 x 371 x 348 mm³ Power: 16 W nominal; Data rate: 5.5 kbit/s raw; 2.7 kbit/s compressed
Overview:	Lid; Trajectory sensor; Time of flight (TOF) mass spectrometer

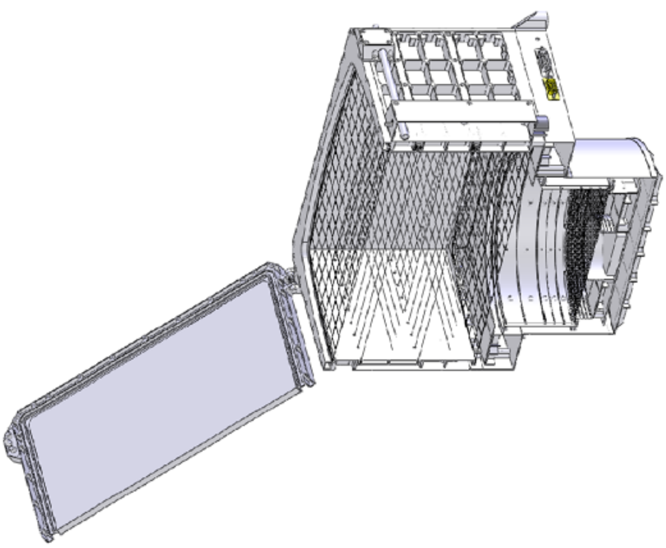


Table 3.1-5: LEVIS Overview

LEVIS (Lunar Exploration Video Imager System)	
Measurement objective:	The LEVIS Camera shall perform high definition video imaging for purposes of public outreach during the mission
Key requirements:	LEVIS shall observe the moon and near environment in constellations, which are not provided by the scientific instrument set, e.g.: – earth rise/set constellations – S/C flight over moon terminator – moon surface with near nadir up to limb pointing – imaging of potentially active regions (e.g. Ina) and potential impact events – provide panoramic view sequences LEVIS shall observe dedicated mission phases/events: – the separation of daughter spacecrafts (AGR) – the impact on moon surface of daughter spacecrafts at end of mission (TBC, if possible by mission) – events during S/C transfer to moon (TBC, if possible by mission)
Main design drivers:	Low mass, low cost, low power High quality high definition video 1 / 2 axis pointing mechanism
Budgets:	Mass: 2.45 kg Envelope: 219 x 170 x 186 mm³ Power: 14 W nominal Data rate: <25 Mbit/s compressed
Overview:	

Table 3.1-6: LunarMag Overview

LunarMag (Lunar Magnetometer)	
Measurement objective:	Global mapping of the lunar vector magnetic field. Measuring magnetic anomalies. Separate the external field from the lunar field.
Key requirements:	The LunarMag experiment shall provide the magnetic field vector in a moon related coordinate system. – The magnetic field measurement shall be performed continuously during the whole mission. – Measurements shall be possible also during the moon eclipse – Synchronous measurements with two sensors on each satellite shall be used to separate remaining spacecraft interference fields. The two sensors (outboard and inboard sensor) shall be mounted on a boom at a distance of about 0.2 * boom length (70cm for a 3.5m long boom) from each other – Synchronous measurements onboard two orbiting sub-satellites shall be used to separate field of solar wind and Earth magneto-tail from moon fields
Main design drivers:	Magnetic cleanliness of the subsatellites Required length of the boom Seperation of the lunar field from the external field requires two sensors on two S/Cs
Budgets:	For one subsatellite: Mass: 1.5 kg Envelope: Boom segment ø100 x 800 mm³; 180 x 120 x 60 mm³ Power: 2.6 W nominal Data rate: 3.38 kbit/s raw
Overview:	Sensor assembly; Electronic box

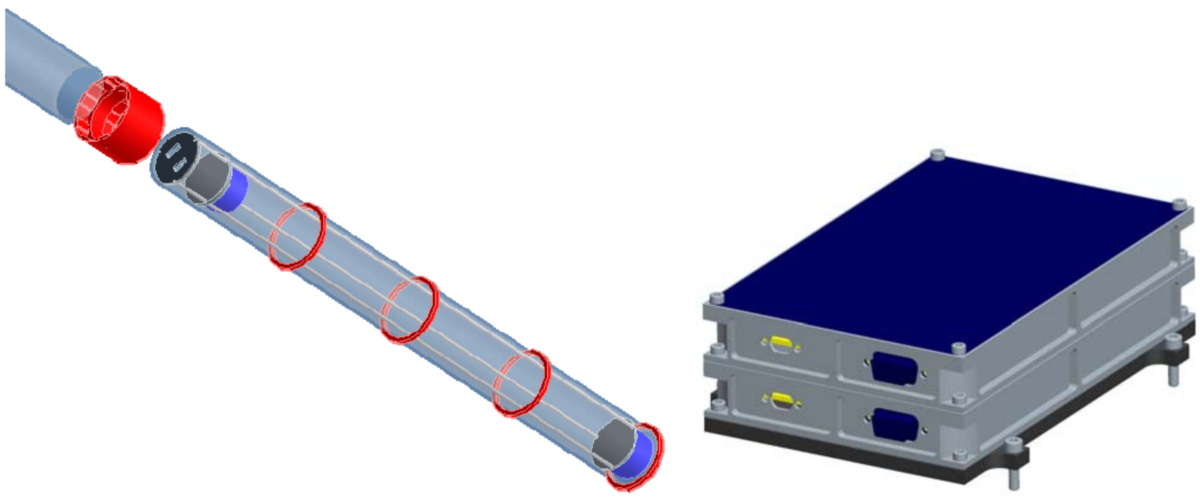
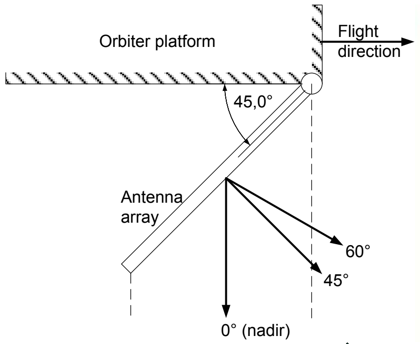


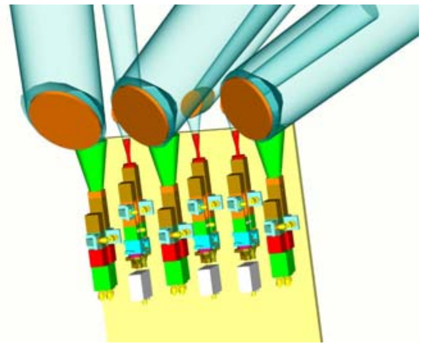
Table 3.1-7: MIMO Overview (1/2)

MIMO (Microwave Instrument for a Moon Orbiter)	
Measurement objective:	– The thermal albedo factor, i.e. the emissivity of particles which form the regolith (from dual polarisation microwave brightnesses). – The conductivity ratio, i.e. the relative impact of IR-radiation and solid state conductivity (from thermophysical modelling and the measured microwave brightness. – The thermal inertia (from thermophysical modelling of the measured lunar microwave brightness). – The dielectrical properties of the medium (from the polarization of the microwave emission and the thermal th), i.e. the refractive index and the microwave opacity. – The temperature and the thermal diffusion
Key requirements:	Regolith mode: – Broad frequency coverage, at least 5 bands: 7, 24, 60,183 and 557 GHz – Three observation angles: 0, 45 and 60 degrees – Vertical and horizontal polarisation – Sensitivity < 0.5 K (brightness temperature) per second integration time – Calibration accuracy: < 0.5 K – Beam width for all frequency 3.5 degrees (HWHM) Limb mode: – Observation of rotational ground state of water near 557 GHz – Spectral Resolution: < 150 KHz – Beamwidth: 0.063 degrees (HWHM) – Appropriate scanning mechanism for azimuth and elevation. Receiver sensitivity < 1500 K DSB (passive cooling required)
Main design drivers:	Measurement of absolute power density (PDF) Thermal stability of antennas difficult esp. for low frequency Spatial resolution drives the size of antenna for lowest frequency Multiple view angles (2/3) required for topographic resolution Wavelength range needs to spread over a factor of 2 to 3 (<5) in order to gain topographic resolution 557 GHz required for water vapour detection in polar regions Limb mode for 557 GHz required for water vapour, no lunar surface within main lobe of antenna, but as close to limb as possible.

Table 3.1-8: MIMO Overview (2/2)

MIMO (Microwave Instrument for a Moon Orbiter)	
Budgets:	Mass: 26.4 kg + TBD kg radiators Envelope: x x mm³; x x mm³ Power: 53,6 W nominal Data rate: 80 kB/s raw; 2 kB/s compressed
Overview:	Microstrip antenna;





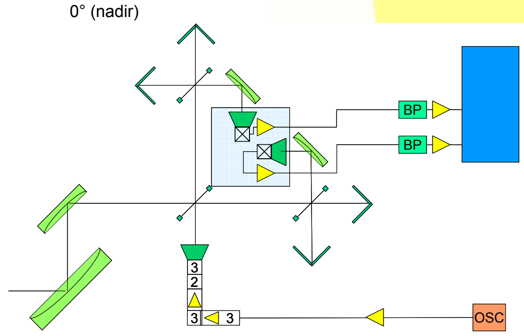


Table 3.1-9: PRARE-L Overview

PRARE-L (Precise Range And Range Rate Equipment - Lunar)	
Measurement objective:	Measure range rate between the subsatellites with high accuracy Measure satellite position in lunar orbit
Key requirements:	Accuracy of range rate measurement: 3 µm/s @ 5s Accuracy of ranging ground to satellite: 1 m Accuracy of time synchronisation: 100 ns (to UTC)
Main design drivers:	Thermal stability of Ka-band frontend Adequate resource demand for subsatellite Thermo-mechanical stability of phase centre of Ka-band antenna to S/C COG Success of gravity experiment depends on many system elements (two subsatellites, two PRARE-L transceivers for intersatellite link, two PRARE-L transceivers for downlink to Earth, PRARE-L ground station)
Budgets:	Mass: 17 kg Envelope: 300 x 240 x 220 mm³; Ka-antenna 100 x 50 x 20 mm³; TT&C 260 x 190 x 50 mm³ Power: 70 W nominal Data rate: 0.25 Mbit/s raw
Overview:	

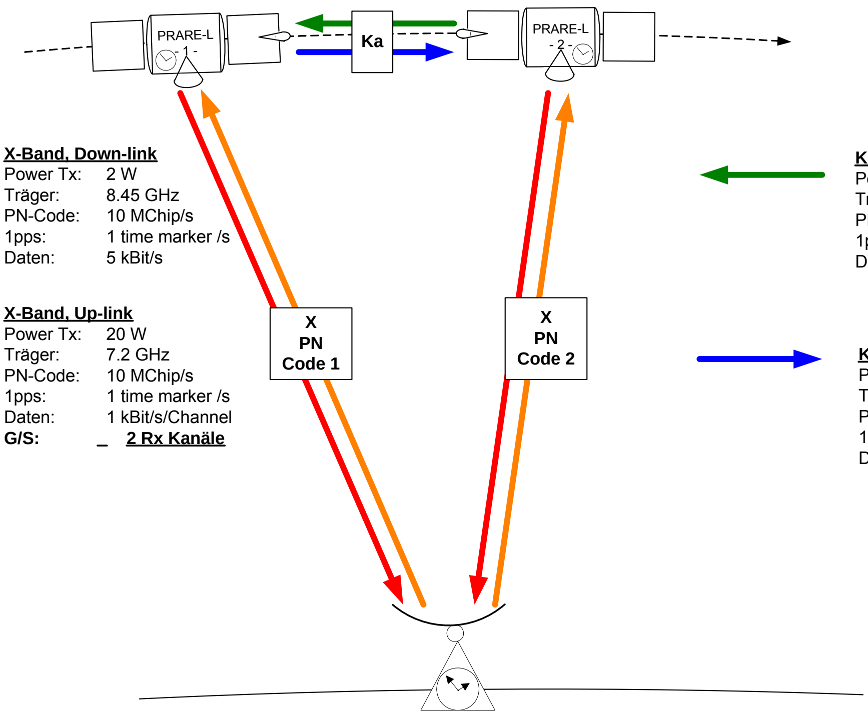


Table 3.1-10: RadMo Overview

RadMo (Radiation Monitor)	
Measurement objective:	– measure ions in the energy range from typically 10 -25 MeV/nuc (this is species dependent) up to 200 MeV/nuc (this is again species dependent) with a resolution of 15%. – measure electrons from 300 keV to 10 MeV (and beyond at much lower confidence) – measure gamma rays from 50 keV to 3 MeV with a resolution of typically 15% – measure the flux of thermal and non-thermal neutrons
Key requirements:	RadMO shall measure particle spectra during Solar Particle Events (SPEs) covering the energy range relevant for astronaut safety and associated science at a time resolution sufficient for accurately determining onset times of SPEs. RadMo shall determine particle composition including neutrons and gammas. RadMo shall establish a global coverage of the radiation environment of the Moon
Main design drivers:	Measurement of SPE particle spectra drives the geometric factor of HET Relevant energy range sets the minimum size of the calorimeter Time resolution < 20s sets geometric factor of HET Measure particle composition drives the HET FOV Detection of neutrons and gamma rays requires HET and NTNS calorimeters Global coverage requires RadMo to be on at (virtually) all times Appropriate data staging ground segment
Budgets:	Mass: 7.451 kg Envelope: HET 150 x 230 x 135 mm³; TNS 150 x 150 x 50 mm³; NTNS 250 x 130 x 130 mm³ Power: 11.4 W nominal Data rate: 116 kbit/s
Overview:	HET (High Energy Telescope); TNS (Thermal Neutron Sensor); NTNS (Non-Thermal Neutron Sensor)

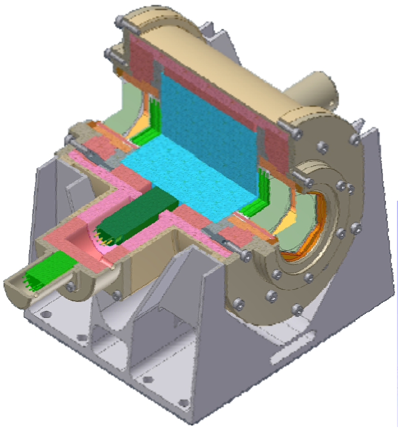
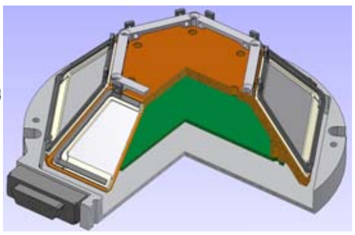
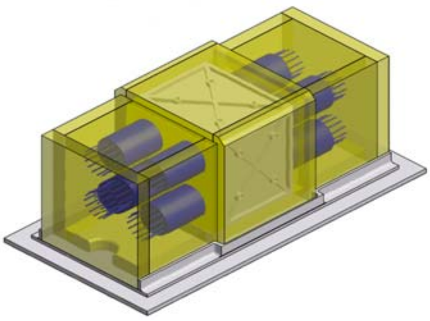




Table 3.1-11: SERTIS Overview

SERTIS (Selenologic Radiometer and Thermal IR Spectrometer)	
Measurement objective:	Measuring the spectral emittance in the spectral range from 7-14µm with an adequate spatial and spectral resolution, especially to measure the Christiansen features and Reststrahlen bands in range 7-10µm. Measuring in the spectral range from 7-40 µm to study the thermo-physical properties of the surface material.
Key requirements:	Spectrometer: Ground Sampling Distance (GSD): < 200 m at 50 km orbit SNR (7 – 10µm): > 100 at surface temperature > 350 K SNR (10 – 14µm): > 50 at surface temperature > 350 K Spectral Sampling Distance (SSD): < 200 nm Dwell time: > 21 ms Radiometer: Ground Sampling Distance (GSD): < 1000 m at 50 km orbit Noise Equivalent Temperature Difference (NETD): < 1K at surface temperature > 100 K
Main design drivers:	GSD; SNR; SSD; NEDT Swath width should be 6 km, nevertheless 3.5 km are considered, since an existing design promises reduced cost and budgets; Global coverage for the reduced swath has to be investigated, but is considered feasible.
Budgets:	Mass: 3.4 kg + TBD kg radiators Envelope: 160 x 181 x 135 mm³ + baffles Power: 15 W nominal; 5 W standby Data rate: 3 Mbit/s raw; 0.4 Mbit/s compressed (max)
Overview:	

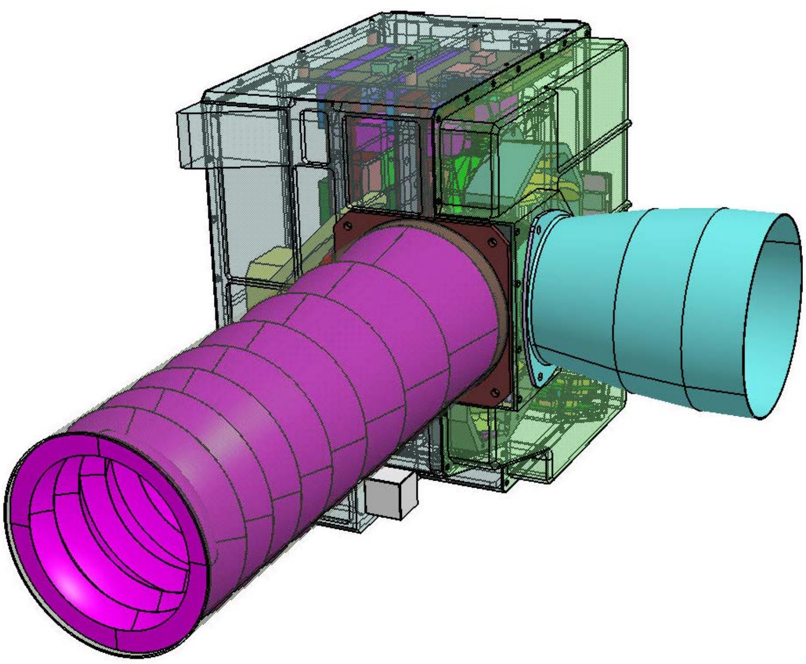


Table 3.1-12: USMI Overview

USMI (Ultraviolet Spectrometer for a Moon Mission)	
Measurement objective:	– Multispectral image data of the lunar surface – Scanning of the lunar surface in the UV in 8 (9 as a goal) spectral band-passes covering the 200 nm – 400 nm wavelength range – Ortho-image colour map generation, showing the albedo of the moon surface in the different spectral band passes.
Key requirements:	The nominal altitude over the moon: 50 km with variation from 25 to 120 km. Within this range USMI shall be able to generate quadratic pixels with TBD accuracy. 8 (9 as a goal) spectral bands from 200 – 400 nm. Definition of spectral domains: TBD CCD lines perpendicular to flight direction <= 0.05 ° (sum of yaw accuracy, misalignment of bus and focal plane) For all spectral channels an SNR of 200 shall be achieved under the following conditions: – Nominal altitude – 5 % minimum moon albedo (TBC) for all wavelengths – Solar incidence angle ± 45 deg The swath width of the instrument shall be 6.144 km with 12 m GSD Pointing Knowledge (yaw axis, a posteriori): 240 +/- 10 µrad Out-of-band photons shall be 1 % of the total of the detected photons over the 200 – 1200 nm spectral range Radiometric resolution: TBD Absolute radiometric accuracy shall be 5 % Integration time (derived requirement): about 7 ms Total mass shall not exceed 10 kg TBC
Main design drivers:	GSD leads to short integration times requiring in combination with SNR a TDI sensor 200 nm channel requires reflective filter design since no transmission filters for this wavelength range exist.
Budgets:	Mass: 5.4 kg + TBD kg radiators Envelope: 470 x 200 x 200 mm³ Power: 33.8 W nominal; 10.14 W standby Data rate: 25.8 Mbit/s raw; 12.9 Mbit/s compressed
Overview:	Two transmission optics; Reflective optics on top

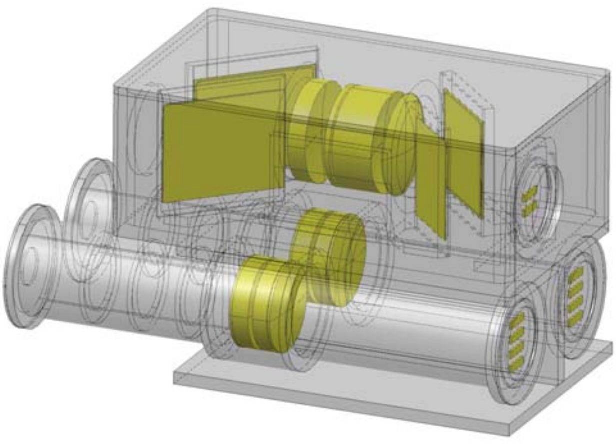


Table 3.1-13: XRF-L Overview

XRF-L (X-Ray Fluorescence - Lunar)	
Measurement objective:	The XRF-L instrument will map concentrations of rock-forming elements (in particular Na, Mg, Al, Si, K, Ca, Ti, Mn, Fe) in high spatial resolution and lunar-wide coverage by using fluorescence of rock-forming elements caused by solar x-ray radiation in the energy range of 0.5-10 keV. This will be achieved by employing an XRF instrument in combination with a direct and an indirect solar monitor.
Key requirements:	XRF-L scans the lunar surface in the x-ray spectral range of 500 eV to 10 keV (2.5-0.12 nm). XRF shall obtain global coverage. XRF-L shall have a spectral resolution of 160 eV @ 6 keV. XRF-L shall have a single pixel yielding a spatial resolution of 6 km (NA) at a reference orbit of 50 km altitude. XRF-L shall achieve an integral count rate of $2 \cdot 10^3$ per pixel. XRF-L shall include two sun monitors, ISM and DSM. XRF-L main instrument detector plane shall operate below a temperature of -20°C. XRF-L instrument parts ISM and DSM shall operate below a temperature of -20°C at their detectors. XRF-L instrument parts ISM and DSM shall independently monitor the solar flux and stay switched on during day time. XRF-L requires a reconstructed pointing knowledge with an uncertainty better than 0.5° and spatial accuracy of better than 500 m in orbit. The integral FOV of the ISM and DSM has to be known with an accuracy of about 10%
Main design drivers:	The required operating temperature of the detectors might require active cooling with severe power demand and large radiators The orbit position knowledge can not be guaranteed, it has to be assessed whether reduced performance is acceptable.
Budgets:	Mass: 12.6 kg + TBD kg radiators Envelope: XRF-L 180 x 360x 100 mm^3; ISM 210 x 60 x 70 mm^3; DSM 40 x 81 x 26 mm^3; OBC 170 x 110 x 25 mm^3; DC/DC-Converter 170 x 110 x 30 mm^3 Power: 30 W nominal Data rate: 2.5 kbit/s
Overview:	XRF-L set up and single element

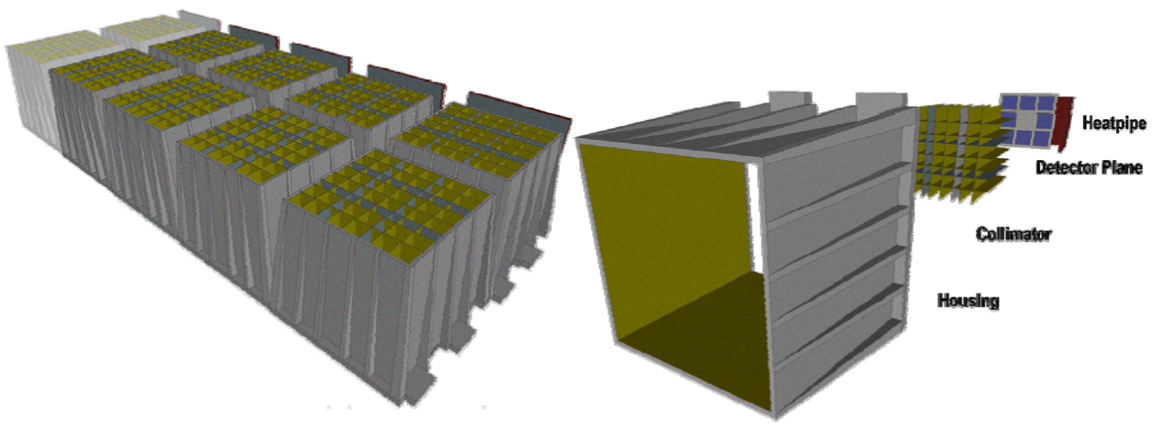


Table 3.1-14: SPOSH-L Overview

SPOSH-L (Smart Panoramic Optical Sensor Head - Lunar)	
Measurement objective:	Detect impact events on the lunar surface during eclipse
Key requirements:	Detection of events brighter than 7m Push broom scanning with array detector 120° x 120° FOV Spectral range 400 - 850 nm
Main design drivers:	Anti-coincidence with cosmic rays remains challenging, possibly dedicated detectors necessary.
Budgets:	Mass: 5 kg incl. margin Envelope: 200 x 200 x 200 mm³ Power: 10 W nominal Data rate: 1.4 kbit/event
Overview:	SPOSH-L-VIS

Table 3.1-15: RaPS Overview

RaPS (Radiation Pressure Sensor)	
Measurement objective:	Determination of the radiation flux distribution
Key requirements:	TBD
Main design drivers:	Sensitivity on night hemisphere
Budgets:	Mass: 6x 0.15 kg Envelope: 6 sensors 95 x 95 x 45 mm³ Power: 0 W nominal Data rate: 20 bit/s
Overview:	RaPS Sensor with sensor segments

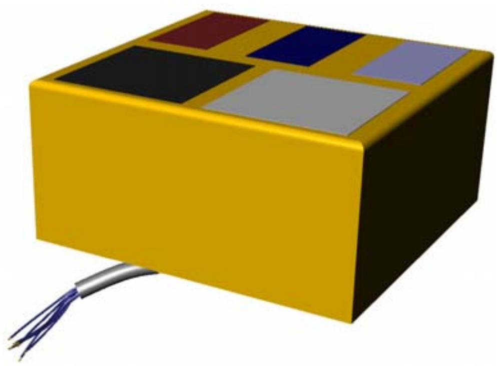
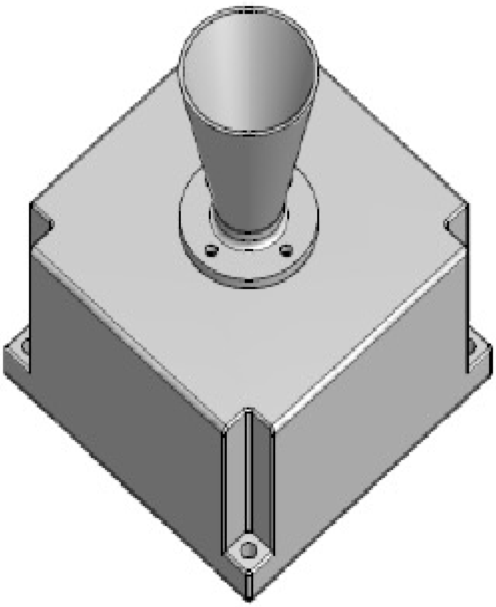


Table 3.1-16: ATON Overview

ATON (Autonomous Optical Navigation)	
Measurement objective:	Qualification of sensor system for autonomous optical navigation over known terrain
Key requirements:	Demonstrate autonomy, accuracy and stability of this new navigational sensor system Gain TRL 7
Main design drivers:	Limited mass and power budget Dynamic range of lunar surface Required accuracy of navigation solution (horizontal :1% of orbit altitude; vertical 0.5% of orbit altitude)
Budgets:	Mass: 1 kg Envelope: 150 x 150 x 100 mm³ Power: 10 W nominal Data rate: 40 Mbit compressed per operation
Overview:	ATON



3.2 Requirements of Instruments on Main Satellite

3.2.1 HRSC-L

The High Resolution Stereo Camera for the Lunar Exploration Orbiter (HRSC-L) is dedicated to generate a comprehensive set of maps from the moon surface with unmatched 3D spatial resolution.

The HRSC-L shall provide imaging data of the lunar surface for analysis of the lunar morphology, topography and chemistry. These data shall allow for:

- Ortho-image map generation with ~1,25 m GSD
- Topographic (stereo) map generation ~4 m sampling (1,25 m GSD for stereo channels required)
- Color map generation with ~2,5 m GSD
- Polarimetric characterization of the lunar surface with ~2,5 m GSD
- Photometric characterization of the lunar surface with ~2,5 m GSD

The instrument shall enable global coverage, respectively twice global coverage with different solar azimuth angles, of the lunar surface with all functions. The instrument shall achieve the specified performance at solar elevation angles from 20° to 60° in regions where such solar elevation angles occur. The dynamic range shall be large enough to cover the dynamic range of the extreme scenes in the polar region.

The HRSC-L shall provide data to analyse and to improve our current knowledge with respect to the:

- Selenodetic control-network
- Global digital terrain model (DTM)
- Lunar surface scattering properties
- Linear polarisation by the lunar surface

The HRSC-L shall provide global coverage for the standard products. Gaps in permanently shadowed areas are tolerable.

HRSC-L shall operate in the following spectral ranges:

- Panchromatic: 490-790 nm
- Blue: 405-480 nm
- Green: 500-560 nm
- NIR1: 720-780 nm
- NIR2: 900-1000 nm

The nominal along- and across-track ground sampling distance of the HRSC-L at the sub-satellite point shall be:

- ≤ 1.25 m for the panchromatic channels
- ≤ 2.5 m for the polarimetric and the colour channels

defined relative to the LEO reference orbit with an altitude of 50 km above the lunar reference sphere.

The nominal minimum vertical resolution of the digital terrain model generated with HRSC-L data shall be ≤ 1.6 m (1.3 times GSD) at the reference orbit altitude of 50 km above the lunar reference sphere.

The stereo mapping function of the HRSC-L shall be implemented with the following 5 stereo angles:

- -18,9° +/- TBD°
- -12,8° +/- TBD°
- 0° (nadir) +/- TBD°
- +12,8° +/- TBD°
- +18,9° +/- TBD°

The different viewing angles will also measure the phase-angle dependent scattering properties of the lunar surface for photometric studies and modelling.

The polarimetric mapping function of the HRSC-L shall be implemented in the panchromatic spectral domain by measuring the linear polarisation parallel and vertical to the flight direction. The nadir-oriented panchromatic channel will be used to derive the required albedo information. The degree of linear polarization shall be measured with a relative accuracy of 10%.

The polarimetric channels shall be accommodated as close as possible to the nadir channel on the focal plane. The off-nadir angle of the four colour channels Blue, Green, NIR1, and NIR2 shall be < 8°.

HRSC-L shall be able to cover the following dynamic range of radiances within one data take:

- Panchromatic: 0.0008 - 0.159 W m-2 sr-1 nm-1
- Panchromatic polarimetric: 0.0008 - 0.159 W m-2 sr-1 nm-1 (TBC)
- Blue: 0.0008 – 0.150 W m-2 sr-1 nm-1
- Green: 0.0011 - 0.156 W m-2 sr-1 nm-1
- NIR1: 0.001 – 0.134 W m-2 sr-1 nm-1
- NIR2: 0.0007 – 0.098 W m-2 sr-1 nm-1

The HRSC-L shall provide a minimum signal to noise ratio (SNR) from an altitude of 50 km above the lunar reference sphere and at full pixel resolution for typical lunar dark material at solar elevation of 10°:

- Panchromatic stereo: ≥ 150
- Panchromatic polarimetric: ≥ 100
- Blue: ≥ 100
- Green: ≥ 150
- NIR1: ≥ 150
- NIR2: ≥ 150

The HRSC-L shall maintain the full dynamic range of the applied detector after A/D conversion.

3.2.2 LEOPARD

The LEOPARD detector shall characterize the moon's dust envelope by measuring mass, speed vector, electrostatic charge, and chemical composition of individual dust grains of the dust cloud surrounding the moon. Therein the LEOPARD instrument shall generate following scientific primary measurement data:

- Mass, speed vector, electrostatic charge, and mass spectrum of the registered grains

From these the following secondary data can be derived by modelling and calculations on earth:

- Reconstructed grain trajectories and their starting points on the Lunar surface
- Density maps of the Lunar dust envelope
- Distribution of the dust production on the Lunar surface

- Maps of the composition of the Lunar regolith

Therefore the measurements from LEOPARD will increase the knowledge about the moon by providing the following models:

- Model for the dust production by hypervelocity impacts on planetary surfaces
- Model of the interplanetary dust flux at 1 AU
- Model for the interstellar dust flux into the solar system

As the LEOPARD detector shall globally characterize the lunar dust cloud, it needs to be in operation for significant parts of the mission. Occasional observation gaps of less than one day are acceptable, since they can be compensated for in the course of a at least 4-year mission. Systematic data gaps longer than one month would reduce the science return significantly and should therefore be avoided.

The LEOPARD detector shall furthermore perform measurements during the LEO cruise phase between earth and moon to acquire interstellar and interplanetary dust data and information about space debris.

As the dust impact speed significantly depends on the orbiting speed of the satellite, the boresight of the LEOPARD detector shall be oriented in spacecraft flight direction. For measuring dust with significant radial components, the FOV of LEOPARD shall not be obscured within ±60° with respect to the instrument boresight.

The LEOPARD instrument shall avoid exposure to Sun light within its FOV, to avoid overheating of its internal electronics. However occasional exposures of less than 3 min (TBC) can be tolerated.

LEOPARD requires to be protected from conducted and radiated EMI generated by other instruments or the spacecraft. EMI charge amplitudes in the range 1 kHz – 100 kHz (TBC) sensed by the trajectory sensor must be smaller than 0.1 fC (TBC). During measurements LEOPARD shall not be exposed to mechanical vibrations between 1 kHz – 100 kHz, as these would cause charge signals on the sense wires of > 0.1 fC.

Knowledge of the electrostatic potential is an essential requirement for determining the size of the detected grains. The potential shall be acquired by a Langmuir probe with at least 0.1 V accuracy and 1 min time resolution at the spacecraft location. Furthermore it shall measure the local number density and temperature of the plasma electrons in the vicinity of the Moon with accuracy better than 10%. The Langmuir probe shall be mounted on a boom of about 1m length with orientation toward the plasma RAM direction.

To reconstruct the ejecta trajectory from the dust speed vector, LEOPARD shall measure the dust speed vector (direction and absolute value) with accuracy better than 0.1% for grains speeds of 2 km/s (the orbital satellite speed). The instrument shall measure the dust mass and the dust charge for grains > 0.1 µm, which the Lunar ejecta cloud is mainly composed of. It shall as well provide TOF mass spectra with a mass resolution better than 100 within a mass range between 1 and 200 u.

For proper mapping, LEOPARD requires knowledge of the spacecraft time with an uncertainty < 0.1 s and a (reconstructed) point with an uncertainty < 0.1°.

3.2.3 LEOSAR

The most important LEOSAR performance requirements are listed in Table 3.2-1: The SAR performance parameters have to be reached for the nominal orbit height of 50 km. The strong orbit variation requires that the radar pulse repetition frequency has to be adapted according to the orbit height above ground. For this purpose a tbd link between the AOCS and the radar is required.

For the SOUNDER mode the performance will be taken as it results from the SAR system design. No special attention will be driven on it, neither will it be in any kind system driving.

The polarisation of LEOSAR is not finally defined during phase A. The LEOSAR instrument will allow for S/W defined polarisation selection: dual or single linear transmit with selectable phase difference and simultaneous dual linear receive. This allows for free selection of the imaging mode – dual linear, dual circular, hybrid polarisation or quad-pol polarisation - even in orbit by appropriate commanding.

The base line operational mode will be transmit in one of the two circular polarisations and simultaneous receive of the two linear ones. This baseline mode is again called dual pol operation.

Table 3.2-1: Key Performance Requirements of the LEOSAR

Parameter	Einheit	Wert	Kommentar
Centre Frequency	GHz	1.27	L-Band
Max Signal Bandwidth	MHz	50	
Orbit Height	km	50	There is a strong variation of the orbit height. The SAR performance is specified for the nominal height only. A link between AOCS system and instrument control is required to enable orbit dependent Radar timing (TBD)
SAR Mode			
Look angle	°	30° - 40°	Off Nadir
Spatial Resolution on ground	m x m	6 x 6	Single look
Swath width	km	12	
Polarisation		Dual and quad pol imaging	Baseline is dual polar imaging. For specific tbd areas quad pol imaging will be applied also
Sensitivity (NESZ)	dB	-25	
Max. Data Rate	Mbps	100	For each of the two Rx channels
Interferometry			
Phase Stability		tbd	
Knowledge of interferometric Basis and Orbit		tbd	It is assumed that posteriory orbit evaluation will give the necessary accuracy. Requirements tbd by DLR
Sounder Mode			
Look angle	°	0	To nadir, S/C roll maneuvre
Spatial Resolution	m x m	10 x 12000	Along x across track
Swath width	km	12	

The quad-pol mode is required for special areas of interest, which will be determined by dual pol observations during the LEO mission. The resulting data rate as well as the required power is in principle doubled compared to the dual pol mode. So the quad pol mode defines the required S/C interfaces power and data rate. Its duration is restricted by the overall data handling philosophy (down link capacity).

Beside the described performance parameters, the over all requirements for

- low mass, ≤ 20 kg for radar electronic + 2 kg for the SAR antenna
- optimised data amount
- low power demand

are system driving and finally determine the selected instrument concept.

3.2.4 MIMO

MIMO will provide a global map about the composition, structure and the physical (electrical and thermal) properties of the lunar regolith down to a depth of ~1 meter (and where appropriate about the thickness of the regolith) and determine the lunar heat flux with higher accuracy than before. Furthermore MIMO will search for water ice and with very high sensitivity for water vapour in the vicinity of polar craters in which up to 4 billion tons of frozen water are believed to be stored. In both cases the method of passive microwave remote sensing will be applied. From the raw data observing the regolith ("regolith mode") the following parameter of thermo-physical models will be derived:

- The thermal albedo factor, i.e. the emissivity of particles which form the regolith (from dual polarisation microwave brightnesses).
- The conductivity ratio, i.e. the relative impact of IR-radiation and solid state conductivity (from thermophysical modelling and the measured microwave brightness.
- The thermal inertia (from thermophysical modelling of the measured lunar microwave brightness).
- The dielectrical properties of the medium (from the polarization of the microwave emission and the thermal), i.e. the refractive index and the microwave opacity.
- The temperature and the thermal diffusion

From these parameters, boundary conditions about the particle size, the elemental and mineralogical composition can be derived.

Furthermore MIMO will search for water in the vicinity of the polar craters. Either water vapour will be directly detected and quantified or upper limits of water will be derived.

<u>Instrumental Key requirements</u>

Radiometer mode (regolith mode):

- Broad frequency coverage, at least 5 bands: 7, 24, 60,183 and 557 GHz
- Three observation angles: 0, 45 and 60 degrees
- Vertical and horizontal polarisation
- Sensitivity < 0.5 K (brightness temperature) per second integration time
- Absolute radiometric accuracy: < 0.5 K
- Beam width for all frequency 3.5 degrees (HWHM)

From lunar samples, obtained during the Apollo missions we know that the penetration depth of microwaves into the lunar regolith is between at least 10 and in some cases larger than 20 wavelengths. Observing in a broad spectral range therefore provides information about the regolith brightness temperature as a function of depth. Additional information, namely about the refractive index is provided by observing the lunar emission in horizontal and vertical polarisation as function of the plane of incidence.

As a trade-off between complexity and scientific benefit the radiometric emission measurement are constrained to 5 spectral bands from 7 to 557 GHz and a minimum of three look / incidence angles in the plane spanned by nadir and direction of flight vectors, in case a continuous scanning over look angle is not reasonable. For each frequency / angle combination both orthogonal polarizations of emission have to be acquired. The discrete look directions of the radiometer antennas are 0° (nadir), 45° and 60° in foresight or backsight.

The geometric resolution along-track is limited by the beam footprint lengths (along) and radiometric integration times. The surface resolution in the orthogonal direction is solely defined by the beam footprint dimension across-track. Depth resolution is provided by the multi-frequency measurement, i.e. each emission band corresponds to a certain penetration depth.

Since the regolith mode is a radiometric measurement mode the only physical parameter to be measured is the brightness temperature in the respective microwave/mm-wave frequency bands. From these measurement parameters the thermal parameters mentioned above (e.g. albedo factor, thermal conductivity) are calculated using thermo-physical models.

Water vapour mode (limb mode):

- Observation of rotational ground state of water near 557 GHz
- Spectral Resolution: < 150 KHz
- Beamwidth: 0.063 degrees (HWHM)
- Appropriate scanning mechanism for azimuth and elevation.
- Receiver sensitivity < 1500 K DSB (passive cooling required)

Within a latitude of 85 degrees pole wards there are a number of craters which may contain water ice, as indicated by neutron spectrometer measurements on Lunar Prospector. During certain illumination conditions part of this water ice may be warmed up and its sublimation rate will increase by orders of magnitude so that water vapour may be detectable. The measurement of the rotational ground state of water vapour is the most sensitive method detecting even smallest amounts of water vapour. We will spectroscopically investigate this water vapour line. The line amplitude will contain information about the amount of water vapour along the line of sight.

3.2.5 RadMo

RadMo shall provide an accurate assessment of the lunar radiation environment, providing data on both the primary galactic cosmic ray (GCR) and solar energetic particle (SEP) component, as well as the induced secondary component, especially neutral particles (neutrons and gammas). Thus, RadMo needs to cover the following functions:

(1) RadMo needs to measure at all times to ensure that solar particle events (SPEs) are observed and to derive orbital and mapped data products.
(2) RadMo needs to be able to discriminate between different particle species.
(3) RadMo needs to be capable of high time resolution (seconds) to resolve onset phases of SPEs and to allow mapping of radiation features to the lunar surface
(4) During quiet times, a slower cadence (20s) allows for mapping the dose across the lunar surface.
(5) To provide accurate estimates of the total equivalent dose, energy deposit needs to be classified according to particle species. This can be achieved by classifying particles, binning and histogramming.

The detailed measurement requirements are given in the System Funtional Specifications document

[AD-01]. In summary, RadMo shall

(1) measure ions in the energy range from typically 10 -25 MeV/nuc (this is species dependent) up to 200 MeV/nuc (this is again species dependent) with a resolution of 15%.
(2) measure electrons from 300 keV to 10 MeV (and beyond at much lower confidence)
(3) measure gamma rays from 50 keV to 3 MeV with a resolution of typically 15%
(4) measure the flux of thermal and non-thermal neutrons

These requirements can not be met with one single sensor. Therefore, RadMo consists of three sensors which cover the above measurement requirements. If possible, all three sensors shall be co-located along an edge of the LEO spacecraft, as near to a corner as feasible to reduce the background that is produced by the interaction of cosmic rays with the spacecraft. If possible, we'd like to be as far from the hydrazine tanks as possible to lower the low-energy neutron background. See [RD1] for drawings. Present accommodation studies show that the RadMo fields of view can be accommodated.

RadMo has moisture- and solvent-sensitive components and, hence, will require dry nitrogen purge up to T-72 hours prior to launch and shall be assembled and tested in a class 100 000 environment. Double bagging shall be used when in less clean environments.

3.2.6 SERTIS

SERTIS shall provide hyperspectral data of the lunar surface in the TIS spectral domain 7 µm – 14 µm that allow to identify feldspar minerals. These can be identified by the location of their Christiansen features as well as by shape and location of transparency and Reststrahlen bands and allow to differentiate feldspars. The instrument shall provide hypersprectral data with a S/N ratio of at least 100 in the wavelength range from 7-10 µm and 50 in the wavelength range from 10-14 µm. The spectral sampling resolution shall be 200 nm or better.

Besides hyperspectal data, SERTIS shall also perform Thermal Infrared Radiometer (TIR) measurements in the wavelength from 7 µm – 40 µm for allowing generation of products of the thermo-physical properties of the lunar regolith. The required NEDT is 1 K at 100 K surface temperature, the nominal spatial resolution for the TIR chanel is ≤ 1 km.

All requirements with priority 1 in the SERTIS chapter of the System Functional Specification LEO-SD-DLR-SP-001 can be regarded as key requirements. These requirements are derived from the following mission goals:

- Determination and cartographic mapping of the lunar surface mineralogy with a high spatial resolution (<200 m ground sampling distance).
- Investigation of the surface composition of the Moon
- Correlation of mineralogy and morphology together with the HRSC-L camera and the VIS-NIR spectrometer
- Study of central peaks and ejecta material of impact craters
- Study the evolution of the lunar volcanism and magma composition
- Study the effects of "Space Weathering" – Comparison with VIS-NIR spectra
- Direct comparison of the mineralogy of Mercury and the Moon
- Global measurement of the surface temperature
- Global mapping of the thermal inertia of the lunar surface
- Global determination of physical regolith properties such as grain size and texture

In 2006-2007 the "Committee on the Scientific Context for the Exploration of the Moon" of the National Research Council of the National Academies prioritized important scientific objectives for NASA. Five out of the 8 high-priority objectives can be addressed with SERTIS, including: 1) Key planetary processes are manifested in the diversity of lunar crustal rocks; 2) The lunar poles are special environments that may bear witness to the volatile flux over the latter part of solar system history; 3) Lunar volcanism provides a window into the thermal and compositional evolution of the Moon; 4) The Moon is an accessible laboratory for studying the impact process on planetary scales; and 5) The Moon is a natural laboratory for regolith processes and weathering on anhydrous airless bodies.

3.2.7 SPOSH

The major requirement that defines the instrument baseline is the present constraint regarding instrument mass to be less than 4kg. This requirement can not be found within the System Functional Specification [AD2]. However, it has been clearly defined during MTP (see MoM MTP).

In order to fulfil this stringent requirement the present baseline cannot reflect all the requirements provided within the SFS. Hereafter we recall some of the major requirements that lead to the definition of the current baseline and have been implemented in the instrument specification.

SP-INT-6.1-200	Instrument mass The SPOSH-L instrument shall have a mass lower than 4kg, excluding typical margins	ALL 1

SP-GEN-5.2-020	Measurement principle and Instrument Architecture The SPOSH-L instrument shall be a pushbroom type, Nadir pointing instrument based on a 2-dimensional CCD detector matrix.	ALL 1
SP-OBS-5.3-030	Observation conditions	ALL
	The SPOSH-L shall monitor the night hemisphere of the lunar surface.	1
SP-OBS-5.3-040	Spectral range	ALL
	SPOSH-L shall monitor the lunar surface in the visible and NIR spectral Range (400nm – 850nm)	1
SP-GEO-5.6-130	Instrument FOV	ALL
	SPOSH-L shall have a viewing angle of ≈ 120° x 120° (170° over the diagonal)	1

SP-RAD-5.5-120	Radiometric noise	ALL
	Assuming illumination equivalent to star magnitude of 7m constant over 0.2s duration SPOSH-L shall provide SNR>5 over the dark hemisphere.	1

3.2.8 USMI

The following key requirements shall apply:

- The nominal altitude over the moon, for which the main requirements apply, shall be 50 km
- Required number of pixels across track shall be 512
- The instrument shall cover nine spectral bands from 200 – 400 nm with typical spectral resolution of λ/10
- For all spectral channels an SNR of 200 shall be achieved under the following conditions:
  - Nominal altitude
  - 5 % minimum moon albedo (TBC) for all wave lengths
  - Solar incidence angle ± 45°
- All filters shall have an integral out-of-band blocking of 1 % over the 200 – 1200 nm spectral range
- The spectral channels shall have an overlap of 0 nm +/− 1 nm.
- The absolute radiometric accuracy shall be 5%.
- The ground sampling distance (GSD) shall be 12 m
- The swath width of the instrument shall be 6.144 km
- The total mass shall not exceed 10 kg TBC

It was discussed whether the shortest wave length channel (200 nm) can be omitted due to specific implementation constraints (see below). For the time being: From scientific point of view it is currently not possible to delete this channel. The necessity for the 200 nm spectral channel shall be investigated during the follow-on study phase

3.2.9 VIS-NIR

The major requirement that drives the instrument baseline is the present requirement on instrument performance combined with the applicable illumination conditions and anticipated spectral resolution. The System Functional Specification [AD2] recalls within the requirement on instrument performance (FR-OBS- 8.2.13-140) an analysis that showed that the required performance levels can be fulfilled for the illumination conditions defined. Unfortunately, details of that analysis are not available. However, a comparable analysis has been performed by JOP within the frame of Phase A activities (see [RD5]). The results of our analysis demonstrate clearly that the required performance is critical for the longwave end of the spectral range. Consequently we suggest to detail and compare both analysis during early phase B.

VN-RAD-5.5-150	Radiometric noise VISNIR shall provide • SNR>150 (TBC) assuming illumination with TBD W/(m²µmsr) representing polar illumination conditions • SNR>400 (TBC) assuming illumination with TBD W/(m²µmsr) representing equatorial illumination conditions within the entire (TBC) spectral range.	ALL 1

VN-GEN-5.2-020	Measurement principle and Instrument Architecture The VISNIR instrument shall be a pushbroom type, Nadir pointing instrument based on a 2-dimensional backthinned MCT detector matrix. The shall be a combination of a TMA (Shafer, TBC) telescope and an Offner-spectrometer	ALL 1

VN-OBS-5.3-030	Observation conditions The VISNIR shall monitor the daytime hemisphere of the lunar surface.	ALL 1

VN-OBS-5.3-040	Spectral range VISNIR shall monitor the lunar surface in the visible, NIR and SWIR spectral range, (400nm – 2300nm). The longwave edge is still TBC. In any case the use of ITAR free off-the-shelf detectors is highly desirable.	ALL 1

VN-OBS-5.3-050	Spectral sampling interval The spectral sampling interval of VISNIR shall be 10nm (TBC)	ALL 1

VN-GEO-5.6-180	Instrument swath Assuming 50km orbit altitude VISNIR shall provide 6km (TBC) swath width at least.	ALL 1

VN-GEO-5.6-190	Spatial sampling distance Assuming 50km orbit altitude VISNIR shall provide 24m (TBC) sampling distance.	ALL 1

VN-INT-6.1-200	Instrument mass The VISNIR instrument shall have a mass lower than 15kg (goal), 20kg (threshold), excluding typical margins	ALL 1

Justification: *Requirement has been defined following the outcomes of the phase A/MTP*

3.2.10 XRF-L

The XRF-L instrument will map concentrations of rock-forming elements (in particular Na, Mg, Al, Si, K, Ca, Ti, Mn, Fe) in high spatial resolution and lunar-wide coverage by using fluorescence of rock-forming elements caused by solar x-ray radiation in the energy range of 0.5-10 keV. This will be achieved by employing an XRF instrument in combination with a direct and an indirect solar monitor. Requirements for operation are listed below:

- XRF-L scans the lunar surface in the x-ray spectral range of 500 eV to 10 keV (2.5-0.12 nm).
- XRF shall obtain global coverage.
- XRF-L shall have a spectral resolution of 160 eV @ 6 keV.
- XRF-L shall have a single pixel yielding a spatial resolution of 6 km (NA) at a reference orbit of 50 km altitude.
- XRF-L shall achieve an integral count rate of $2 \cdot 10^3$ per pixel.
- XRF-L shall include two sun monitors, ISM and DSM.
- XRF-L main instrument part shall point nadir.
- XRF-L main instrument part shall have a free FOV 20° along-track and across-track.
- XRF-L instrument parts ISM and DSM shall point zenith (anti-nadir).
- XRF-L instrument parts ISM and DSM shall have a FOV of 1.5 π.
- XRF-L main instrument detector plane shall operate below a temperature of -20°C.
- XRF-L instrument parts ISM and DSM shall operate below a temperature of -20°C at their detectors.
- XRF-L shall be switched on, while the S/C nadir-face points towards illuminated lunar surface (solar elevation angles 0° - 90°).
- XRF-L instrument parts ISM and DSM shall independently monitor the solar flux and stay switched on during day time.
- XRF-L shall receive a time-signal 1PPS.
- XRF-L requires a reconstructed pointing knowledge with an uncertainty better than 0.5° and spatial accuracy of better than 500 m in orbit.

- The integral FOV of the ISM and DSM has to be known with an accuracy of about 10%

3.2.11 ATON

Recently proposed lunar missions e.g. research at predetermined local targets or the construction of a lunar base require an autonomous, precise, safe and soft landing. For a better understanding of the goal of ATON in the following a short explanation of a lunar landing manoeuvre shall be given.

To perform a landing as stated above, the spacecraft usually starts from an initial orbit around the Moon. The landing commences with a de-orbit burn of ≈ 20-30 m/s, to bring the lander into the descent orbit with a Periselenium of about 10-15 km. At the Periselenium, the powered descent is initiated. The powered descent reduces most of the moon relative velocity of the lander. This manoeuvre takes a Δv in the order of 1700 m/s. The powered descent ends at an altitude in the range of 500 – 1000 m with a remaining spacecraft velocity in the order of 10s of m/s. In the last part of the landing the spacecraft identifies a safe landing site and performs a soft touchdown. The whole landing procedure can be segmented in three phases:

- Phase 1: Transition from initial orbit into descent orbit, lasts until initiation of powered descent
- Phase 2: Powered descent
- Phase 3: Safe and soft landing

For a precise landing, an accurate knowledge of the spacecraft state is necessary, to determine the optimum point for the initiation of the powered descent. This knowledge must be maintained also during phase 2 and phase 3. Additionally in the latter phase the terrain has to be assessed in terms of safety. Whereas there are different approaches in the European space community to handle the guidance, navigation and control (GNC) problems in phases 2 and 3, the navigation problem of the determination of the vehicles state in phase 1 is still unsolved.

The objective of ATON is the demonstration of autonomous, immediate and continuous absolute position determination of a spacecraft in lunar orbit – thus to solve the navigation problem of phase 1 (Req. AT-FUN-5.1-010, AT-FUN-5.1-020). The approach is to navigate by the use of known landmarks situated in the spacecrafts underlying surface. In first place craters are chosen as landmarks.

The strategy of the determination of the state is to take optical measurement (Req. AT-FUN-5.2-010) in the form of images (Req. AT-FUN-5.1-030) of the spacecraft's underlying lunar surface by ATON. After image processing in ATON a certain fraction of the craters in each image is extracted. For each extracted crater its shape and the position of its centre in the image are determined. Additionally the viewing angle is determined. With a sufficient number of craters in one image, their centres form a unique pattern, comparable to a unique star constellation. The identification takes place via the matching of the crater constellation with an on-board data base. The on-board database must contain the position of the craters on the Moon, their shape and their size. After landmark identification the spacecrafts position in space can be determined by comparing the size and shape of the craters in the image to their true values in the data base.

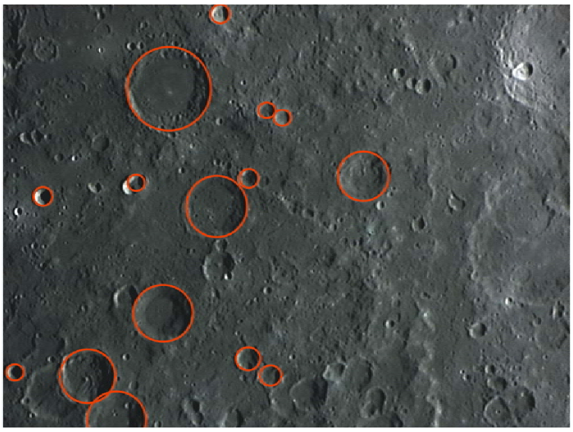


Figure 3.2-1 Extraction of craters

Consequently, this data base must be known in advance and stored in ATON prior to launch. The resulting navigation-fix correlates with its accuracy. The intended accuracy of the database is in the order of tens of meters. Considering a resolution of 1 MPixel and a FOV of 35°, suitable crater diameters are in the range from 600 m to 15 km and higher.

The identification process can be much accelerated by an estimated knowledge of the vehicles position in the order of tens of kilometres. This accuracy should be delivered by the nominal GNC solution of the spacecraft. Considering that, the crater-searchspace reduces to a number in the order of 1000 craters.

ATON shall achieve a positioning accuracy in the order of 100 m (Req. AT-Spe-6.1-010). With the demonstration of this technology in an operational environment a Technology Readiness Level (TRL) of 7 shall be reached (Req. AT-FUN-5.1-050). In order to get sufficient opportunities to test the technology and in order to have the possibility to check the long term behaviour of the instrument a lifetime of 4 years is foreseen (Req. AT-FUN-5.1-040).

The key system requirements of ATON are the needed pointing angle to the lunar surface of +/-30 deg with an attitude stability of 0.3 deg/s (Req. AT-FUN-5.4-010) and an interface to the bus in order to receive AOCS data (Req. AT-Spe-8.2-010). ATON shall conduct at least 50 short term operations (Req. FR-OBS-8.2.15-090) which shall have a minimum duration of 300 seconds of continuous operation (Req. AT-OPS-9.1-010). The operations shall be preferably performed between Aposelenium and Periselenium.

3.2.12 LEVIS

LEVIS (Lunar Exploration Video Imager System) is a high definition digital video camera system for the LEO Mission. It will support the mission public outreach by taking live videos of interesting mission scenarios in high quality (HDTV). Its imaging sequences and capabilities shall be complementary to the scientific instrument set.

The LEVIS camera shall provide High Definition videos, i.e. minimum 720p, typical 1080p Video format, for public outreach.

Therefore the camera has to observe objects of interest, e.g. S/C daughter separation, earth rise/set constellations etc.

These objects lay in different directions, so the pointing has to be adapted, as well as the FOV, because the objects are in rather different range and are of different size.

3.3 Requirements of Instruments on Subsatellites

3.3.1 LunarMag

The necessary measurements need to be performed on a quasi-polar orbit to ensure global coverage of the lunar surface. As the crustal magnetic fields are weak the altitude of the spacecraft should not be larger than 70 km.

The LunarMag experiment shall provide the magnetic field vector in a moon related coordinate system.

- The magnetic field measurement shall be performed continuously during the whole mission.
- Measurements shall be possible also during the moon eclipse
- Synchronous measurements with two sensors on each satellite shall be used to separate remaining spacecraft interference fields. The two sensors (outboard and inboard sensor) shall be mounted on a boom at a distance of about 0.2 * boom length (70cm for a 3.5m long boom) from each other
- Synchronous measurements onboard two orbiting sub-satellites shall be used to separate field of solar wind and Earth magneto-tail from moon fields

Time as well as spacecraft position and orientation shall be provided by the spacecraft operation team

3.3.2 PRARE-L

3.3.2.1 PRARE-L Measurement Principle

The precise knowledge of the lunar gravity field in terms of accuracy and resolution is crucial for understanding the internal structure and evolution of the Moon [RD16]. All currently available lunar gravity field models are mainly based on the analysis of insufficient radio tracking data of orbiting spacecraft, which include data from the Lunar Orbiter, Apollo, Clementine, and Lunar Prospector missions [RD11]. Additionally, due to the synchronous rotation of the Moon around the Earth, all derived models lack information from the far-side. The most recent and probably best model available is the JPL Lunar Prospector mission result LP150Q [RD10] which is complete to degree and order 150 of a spherical harmonic representation. The model can resolve small-scale features up to 36 km half-wavelength with an approximate accuracy of about 30 mGal on the near-side, but, due to the inhomogeneous input data which require strong constraints in the solution process, the model errors are definitely much larger (up to 200 mGal) on the far-side [RD15].

A high accuracy and high resolution lunar gravity field with a global mean error of less than 1 mGal for half-wavelengths of 50 km (which was set as the science requirement for ESA's Moon Orbiting Observatory (MORO) mission [RD8]) would allow to precisely

- determine the structure of the lunar crust by inversion of topography and gravity information,

- investigate the heterogeneity and elasticity of the lunar mantle,
- investigate the reason for the 1.9-km shift between the centre of mass and centre of figure of the Moon,
- improve the characterization of the size, physical state and composition of the lunar core or
- determine lunar Love numbers, which account for the tidal distortion of the Moon, for the first time precisely from time-varying gravitational perturbations

as well as to

- draw conclusions on the origin of the Moon and early Earth history,
- improve the orbit determination and maintenance of the LEO and all other historic and future lunar missions,
- investigate the Earth-Moon system as a whole and in detail the librations of the Moon, or
- conduct relativistic studies in the Earth-Moon system.

The potential of a body (planet or moon) is the exterior potential of the body. E.g, for the Earth system, this includes its entire solid and fluid (including oceans and atmosphere) components. Following conventional methods [RD9], at a field point P, exterior to the body system, the potential of gravitational attraction between a unit mass and the body system may be represented using an infinite spherical harmonic series. The field point P is specified by its geocentric radius *r*, geographic latitude φ, and longitude λ. If μ represents the gravitational constant of the body, and *R* represents its mean equatorial radius (or a scale distance), then the body's exterior potential *V* can be represented as

Equ. 3.4.2.1-1: $$V(r,\varphi,\lambda;t)=\frac{\mu}{r}+\frac{\mu}{r}\sum_{l=2}^{N_{\max}}\left(\frac{R}{r}\right)^{l}\sum_{m=0}^{l}\overline{P}_{lm}(\sin\varphi)\left\{\overline{C}_{lm}(t)\cos m\lambda+\overline{S}_{lm}(t)\sin m\lambda\right\}$$

In this expression, $\overline{P}_{lm}(\sin\varphi)$ are the (fully-normalized) associated Legendre polynomials of degree *l* and order *m*; and $\overline{C}_{lm}$ and $\overline{S}_{lm}$ are the (fully-normalized) spherical harmonic coefficients of the geopotential.

The geopotential at a fixed location may be variable in time due to mass movement and exchange between the body system components (see planet Earth). This is reflected by introducing the independent variable time (*t*) on the left; and is implemented or realized by treating the spherical harmonic coefficients of the geopotential as time dependent. The continuum of variations of the geopotential is represented by theoretically continuous variation of the geopotential coefficients. For the lunar case, the spherical harmonics can be treated as time-independent or static.

The spherical harmonic expansion of the geopotential requires an infinite series of harmonics, practicality dictates that the summation on the right be limited to a maximum degree N_{max}.

Conventionally, the origin of the reference frame maybe chosen to be coincident with the center of mass of the entire body system, including its solid component and fluid envelopes. In this convention, the potential has no terms of degree *l=1* on the right hand side of Equ. 3.4.2.1-1.

The global gravity field of a planet or its moon can principally be derived by three measurement principles:

(1) High-low satellite-to-satellite tracking (HLSST): By this technique the long-wavelength features of the gravity field are determined from tracking between a satellite which is flying on a low-altitude and highly-inclined (to avoid polar gaps which would cause instabilities solving global spherical harmonics) orbit and a 2nd satellite on a higher orbit or a ground

station on the Earth. Typical examples for HLSST Earth gravity field determination are the CHAMP mission (CHAallenging Mini-satellite Payload [RD12]) to improve the Earth's gravity field and for the Moon the above mentioned APOLLO, Clementine and Lunar Prospector missions as well as the recently launched SELENE (SELenological and ENgineering Explorer) mission.

(2) Low-low satellite-to-satellite tracking (LLSST): Here, highly-precise range and/or range-rate observations are performed between a pair of satellites which orbit the planet on a low-altitude and highly-inclined orbit. The LLSST observations enable to derive the mid to short wavelength features of the gravity field. To geolocate the LLSST observations and to determine the large-scale features additional HLSST is required. A good example is the Gravity Recovery and Climate Experiment [RD14] which monitors monthly and even sub-monthly mass variations in the Earth system since 2002 based on K-band SST and the Global Positioning System.

(3) Satellite gravity gradiometry (SGG): This measurement principle combines the above methods, but instead of measuring the range and/or range-rate between a pair of satellites the gravity gradient is directly measured with a set of three pairs of orthogonally implemented accelerometers which enables highly-precise measurements of the low-wavelength gravity field features. As for the LLSST case, additional HLSST is required. This principle will be performed for the very first time on the Gravity and steady-state Ocean Circulation Explorer (GOCE) due for launch in spring 2009.

For the LEO mission LLSST with a ground station on the Earth is the method of choice because a) HLSST is already performed by SELENE to improve the long-wavelength knowledge of the lunar gravity field on a global scale and with homogeneous accuracy (e.g. also on the far side) and another HLSST experiment would have only marginal power for further improvements and b) highly-precise, low-cost and low-weight SGG instruments are not available so far. Additionally, range-rate LLSST measurements are preferred because they are more closely related to gravity gradients than range measurements. Also, instead of artificial range-rate observations, obtained by appropriate range derivation/filter techniques, direct velocity measurements are preferred to minimize numerical errors. The LLSST Ka-band link in combination with X-band HLSST between the LEO satellites and a ground station on Earth for the PRARE-L instrument is depicted in Figure 3.3-1.

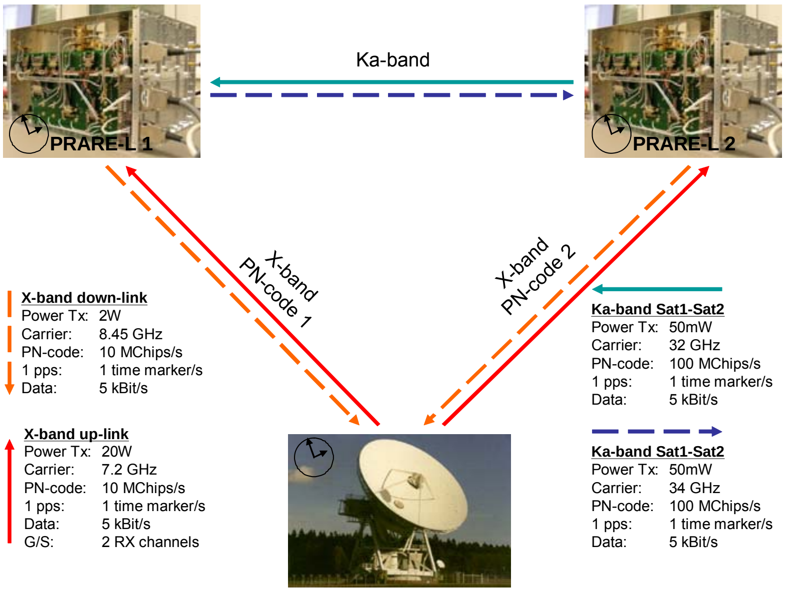


Figure 3.3-1: Current PRARE-L LLSST and HLSST signal link design (showing the ACES-MWL Engineering unit and the Weilheim 30m X-band receive antenna.

Usually the LLSST and HLSST tracking data are exploited during ground data processing using the classical dynamical approach including numerical integration, differential orbit correction and parameter adjustment (see for the GRACE mission [RD13]).

A LLSST mission has to fulfil certain requirements:

- The separation distance *s* between the satellites has to be roughly adjusted to the minimum half-wavelength *λ* or the spherical harmonic degree *n* which should be determined by analysis of tracking data within a certain time span. Here, the equation s = λ = R·π/n can be applied. As an example, in case of the Moon (R=1738 km) a gravity field up to approximately degree and order n = 90 or 60 km half-wavelength can be derived with a satellite separation of roughly 60 km. To guarantee a homogeneous data coverage, the equatorial ground track spacing (which is related to the spherical harmonic order *m*) should have an equal or better spatial resolution. For a perfect LEO repeat orbit within 28 days (resulting in 16*28 revolutions) we will get about 24 km, indicating the power to resolve even smaller gravity field features with just 4 weeks of LLSST. On the other side, a non-repeat orbit would require a longer time span to obtain the desired gravity field resolution.
- The orbital parameters have to be such that a) the orbital height is as low as possible to avoid attenuation of the gravity signal with altitude and b) the inclination should be as high as possible to avoid data gaps in the Polar regions.
- The LLSST data accuracy has to be adjusted to the aimed at mid- to short-wavelength accuracy of the gravity field which is more or less directly proportional to the LLSST data accuracy (experience from GRACE and LEO real data and simulations). In order to get a factor of 1000 for improvement of the mid wavelengths (w.r.t SELENE), an accuracy of some µm/s is required.
- The HLSST data accuracy has to be adjusted to the aimed at long-wavelength accuracy of the gravity field. In order to get a factor of 10 for improvement of the long wavelengths (w.r.t SELENE), an accuracy of better than 1 m is required.
- LLSST and HLSST data gaps (e.g. by manoeuvres, satellite anomalies or ground station maintenance) should be minimized to guarantee a homogeneous and precise orbit and gravity field determination. Also, geographically correlated data gaps shall be avoided (which is of course not possible for the far side of the Moon using HLSST tracking from the Earth).

- The satellites have to be built and operated in such a way that range or range-rate errors due to thermal expansion, multipath, fuel sloshing, movable parts or attitude errors, which all have to be taken into account for transformation of the observation from the LLSST instrument phase centres to the satellite centres of mass, are minimized (e.g. in the order of the LLSST accuracy).
- Non-gravitational forces have to be observed (e.g by accelerometers (Earth case) or by the RaPS instrument (Moon)) or modelled.

Further details on the PRARE-L lunar gravity field determination principle and simulation results can be found in [RD7].

3.3.2.2 Key Requirements of PRARE-L

The PRARE-L system as a central part of the gravity field experiment shall measure the intersatellite range-rate data between the subsatellites and the satellite-to-ground range and range rate data which is required to determine the lunar gravity field with the envisaged accuracy of 0.1 mGal at half-wavelenght of 50 km. Therefore the system has to:

- measure the satellite range-rate data with an accuracy of 3 µm/s @ 5 s integration time between the COGs of the subsatellites
- operate at the nominal relative distance between the subsatellites of 60 km to 120 km
- measure the range between the ground station's antenna phase-centre and the antenna phase centre of the subsatellites with a ranging stability of 10 cm @ 5s and a range-rate stability of 17 µm/s @ 5 s
- synchronise the clocks on ground to space with an accuracy of 50 ns (1σ)
- synchronise the ground reference clock with an accuracy of 100 ns (1σ) to UTC
- synchronise the clocks on both subsatellites to 50 ns (1σ)
- operate the inter-satellite link continuously, data gaps need to be less than 5%
- operate the satellite-to-ground link during visibility of PRARE-L G/S
- provide data block duration of 14 to 28 days of uninterrupted operation

Additional to the science related requirements the PRARE-L system as an integral part of the subsatellites system architecture has to provide specific spacecraft related functions. Beeing crucial for the operation of the satellites a dedicated reliability requirement will have to be put to these functions. The PRARE-L system has to provide the following satellite operation functions:

- operational tracking, also for safe mode, to ensure operability and orbit maintenance
- TM/TC and data downlink for each subsatellite
  - the TM/TC shall be transparent between the onboard computer and the GSOC ground system
  - the TM/TC shall provide safemode functionality

Besides these functional requirements the following design constraints shall be maintained:

- minimise the required subsatellite resources
- optimisation of the thermal stability
- optimise the location of the inter-satellite Ka-band antenna w.r.t. the subsatellite's COG
- stabilise Ka-band antenna's phase centre wrt to the signal incidence angle

It should be kept in mind that the gravity field experiment can only be successful when all elements (two subsatellites, two PRARE-L transceivers for intersatellite link, two PRARE-L transceivers for downlink to Earth, PRARE-L ground station) are functional. The large number of elements puts an issue on the reliability of the overall system.

3.3.3 RaPS

The global spatial and temporal variation of the lunar albedo and infrared field is not known with sufficient accuracy. Especially on the farside no information is available in the current models. Additionally, the radiation pressure, which is the sum of the direct solar radiation and the albedo and infrared radiation of the Moon, acts as a non-gravitational force on the LEO identical sub-satellites (LSS) and needs to be taken into account for lunar gravity field determination. Finally, the degradation of the optical and thermal characteristics of LSS surface areas during mission lifetime is not known, but has significant influence on the radiation pressure force.

Hereby, RaPS measures the incoming radiation flux and the degradation of the thermal and optical characteristics of all important LSS surface areas. The radiation pressure is then calculated with the measured thermal flux the known satellite design and the measured surface areas characteristics degradation.

Additionally, the a-posteriori separation of the lunar albedo and infrared components from the Earth infrared and albedo contribution and from the direct solar radiation will enable studies of the magnetic field of the lunar crust by analysis of the thermal variations of the lunar surface (LUNARMAG synergy).

The RaPS derived data will be generated at a coarse resolution (compared to SERTIS and MIMO derived data). Its advantage however is that it is generated during all seasons and daytimes and from the integral of all radiated wavelengths. Thus RaPS can be used to correlate between the other measurements and to extrapolate these measurements during their off-times.

One main goal of RaPS is to improve the PRARE-L gravity measurement performance. Possibly it can amend the results of other TBD instruments in addition.

The RaPS instrument is based on the foreseen CMSS safe-mode sensor on the LSS that needs to be adapted with only small influence on the system budgets. [AD2] general requirements are anyhow applicable to the safe-mode sensor and do not impose additional design or cost drivers to the RaPS instrument. All RaPS specific requirements detailed in [AD2] are taken into account in the latest RaPS design. None of them is considered to be a significant design or cost driver. Also, the requirements FR-OBS-8.2.9-010 and FR-OBS-8.2.9-020 on the measurement accuracy did not proof to be difficult to meet in the first assessments but will have to be further and closer analysed in the following project phases.

4 Mission and System Description

4.1 Overview

4.1.1 Mission overview

LEO is the first German interplanetary mission. The goal of LEO is the global and multispectral investigation of the Moon. The LEO Space Segment consists out of one Main Satellite and two Sub-satellites. The Sub-satellites are carried to the Moon by the Main Satellite and are deployed by it into their nominal lunar orbit.

LEO will be launched on a Soyuz 2-1b launch vehicle with Fregat-M upper stage from CSG. The Fregat will insert the Space Segment into a direct trajectory to the Moon. The Main Satellite will use the on-board propulsion system to capture into lunar orbit.

After Main Satellite commissioning, the Space Segment will first enter into the Sub-satellites Nominal Orbit of 50 km average altitude and 85° inclination. There the Main Satellite will deploy the Sub-satellites. The Sub-satellites will carry out a nominal mission of 4 years. After completion of the nominal mission, the Sub-satellites may continue operations as part of an extended mission phase on the same orbit. After the Sub-satellites use all of the onboard fuel, their orbits will degrade and they will eventually impact the surface of the Moon.

After the deployment of the Sub-satellites, the Main Satellite will enter its Nominal Orbit 1, which shares the basic characteristics of the Sub-satellites Nominal Orbit. After 3 years in its Nominal Orbit 1, the Main Satellite will carry out an inclination change to the polar Nominal Orbit 2 in which it will operate for 1 year. After completion of the nominal mission, the LEO Main Satellite may continue operations as part of an extended mission phase. The duration of the extended mission phase is dependent on the orbit. After the LEO Main Satellite uses all of the onboard fuel, its orbit will degrade and eventually impacts the surface of the Moon.

4.1.2 System Segments

The LEO System consists of three system segments, as illustrated in Figure 4.1-1.

The Space Segment consists out of three satellites, the Main Satellite and two Sub-satellites. The Main Satellite consists out of the spacecraft Platform and the Payload Suite, and the Payload Support System. The Main Satellite Platform provides the power generation, platform command and data handling, guidance, navigation and control, and the thermal and mechanical systems to support the satellite instruments and the Payload Support System. The Payload Suite consists out of 10 instruments, the technology demonstration payload and the auxiliary payload. The Payload Support System provides the power distribution for the Payload Suite, and the Payload command and data handling.

The two Sub-satellites are identical in design. They consist out of the Sub-satellite Platform and the Sub-satellite Payload Suite. Sub-satellite Platform provides the power generation, command and data handling, guidance, navigation and control, thermal and mechanical systems to support the Sub-satellite instruments. The Sub-satellite Payload Suite consists out of three instruments. The PRARE-L payload onboard the subsatellites has a special role because on the one hand it performs the range-rate measurement for the gravity field determination and on the other hand it serves as the telecommunications system for the subsatellites.

The launch segment is responsible for providing the Soyuz launch vehicle and the corresponding support services. Also contained in the launch segment are the processing facilities and support functions required once the Space Segment arrives at the launch site.

The third and final segment is the ground segment, which consists out of the Mission Operations System and the Science Data System. The LEO Mission Operations System deploys the network of ground stations to communicate with the LEO space segment, provides the operations centre for the mission implements the communication and data networks between ground elements, provides the Flight Dynamics System (FDS) and operates the mission. The Science Data System consists out of the individual Science Operations Centres (SOCs) and the Payload Ground System (PGS). The SOCs receive the instrument data and other mission products used in data product generation. The SOCs will deliver the data products to the PGS for archiving.

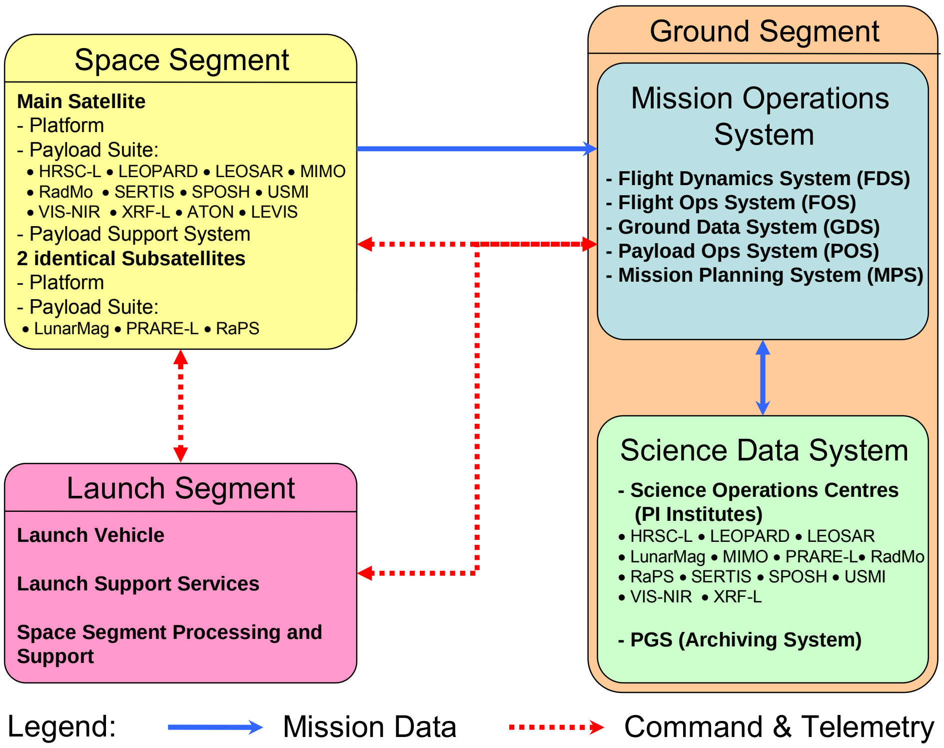


Figure 4.1-1 The LEO mission segments

4.2 Launcher

One major item of the Phase A to be analysed and jointly decided was the decision on the baseline launcher as well as whether to design the mission, to be compliant with several or only one launcher.

The drawback on the mission and system requirements of not having a back-up launcher can be such that a later switch to another launcher is extremely costly and time-consuming (if possible at all). For example, only a few launchers are capable of a direct injection into the Lunar Transfer Orbit (LTO). The question of the need of an alternative launch option is mainly driven by two aspects:

- to have a technical backup and increase flexibility
- to avoid major delays caused by the launch system

As with any other backup or insurance issue, the decision on the backup option is to be decided by the customer. This decision process was finalised for LEO Phase A Mid-Term Presentation with the result:

- Soyuz Fregat from Kourou is selected as Baseline launcher, compatibility of LEO has to be ensured and maintained
- Zenith 3/ LandLaunch shall be a full, nominal backup, compatibility of LEO has to be ensured and maintained
- Falcon 9 should be backup alternative, compatibility of LEO is aimed and tracked. If Falcon 9 compatibility gets design or cost driving, the need for this alternative will be re-assessed.

4.2.1 Potential LEO Launchers

The potential launchers are analysed with respect to availability, compatibility and compliance to the LEO Mission requirements. Technical and Program Criteria for the carrier selection were defined, and cover:

- Sufficient payload capacity for the LEO mission
- Availability for launch in the year 2012
- Proven reliability expected by 2012 after several successful starts
- Flexibility concerning the launch window
- Political aspects (e.g. ITAR)
- ROM costs of not more than 60 M€

Table 4.2-1 summarizes the result of the launcher analysis for LEO. The table shows the performance for the different launch scenarios, with in blue colour the LEO Phase A baseline LTO option.

Table 4.2-1 Potential Launchers for the LEO Mission

Launcher Recommendation for the LEO Mission						
Launcher	**Payload Capacity GTO [1]) [kg]**	**Payload Capacity LTO [kg]**	**Payload Capacity HEO [kg]**	**Fairing Dia. [m]**	**ROM Cost [M€]**	**Status/ Maiden Flight**
Baseline Launcher Recommendation						
Soyuz/Fregat (Kourou) [2])	2930	2150	2060	3.8	50	not operational/ planned Q2 2009
Backup Launcher Recommendation						
Zenit-3SLB	5000	3800	3600	3.6	45	operational/ 28.04.2008
Further potential, but restricted Backup Launchers						
Falcon 9	4600	2500	-	4.6	30	not operational/ planned Q3 2010

As baseline for the LEO Phase A, a launch on Soyuz Fregat is recommended. In summary, the recommended launch systems offer following major aspects to LEO:

<u>Soyuz Fregat (Kourou) – LTO or GTO</u>

- Compliant with ESA launcher policy: European Launcher with German participation from the European spaceport in Kourou
- Technical compliance to the LEO requirements
- Moderate launch cost
- Extensive Launch Record of Launch vehicle, even on LTO launches
- Missing Heritage and no proven reliability from Kourou

<u>Zenith 3SLB (LandLaunch) – LTO or GTO</u>

- Technical compliance to the LEO requirements and extensive margin on needed performance (Status Phase A)
- Moderate launch cost
- Extensive Launch Record of Launch vehicle Zenith and BlockDM upper stage
- Low Heritage and only once proven reliability from Baikonour

<u>Cost Option : Falcon 9 – LTO or GTO</u>

- Very low launch cost – **<u>Cost Option</u>**
- Major risk of launcher development schedule, availability and reliability
- Launcher interfaces are not fully defined and partly unknown today
- No 1194mm standard adapter foreseen today

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[1]) For the LEO mission optimized GTO

[2]) GTO orbital inclination of 23 °

4.3 Mission Analysis

This section covers the mission analysis performed for the LEO mission. It is split into the analysis for the transfer and lunar orbit insertion and into the analysis for the science orbit.

4.3.1 Transfer and Lunar Orbit Insertion

A detailed analysis of the transfer to the moon has been performed in the context of [RD17]. The selected baseline is a direct transfer and lunar orbit insertion. Concerning the launch window, there exists a launch opportunity of 30 min per day for a direct injection into the lunar transfer orbit. However, for a Soyuz launcher, this is constrained to two out of four weeks during a month due to range safety constraints for the impact of the Soyuz 3rd stage.

A direct lunar transfer normally takes between 4 and 6 days. In the baseline concept, three trajectory correction manoeuvres (TCMs) with a total ΔV of 140 m/s are foreseen:

- TCM1 Correction of launcher and Lunar Transfer Injection (LTI) dispersion
- TCM2 Error correction of TCM1
- TCM3 Error correction of TCM2 and preparation of Lunar Orbit Injection (LOI)

A reduction to a total of 40 m/s may be possible with updated data of the Soyuz injection accuracy. A depiction of the transfer orbit and the lunar orbit insertion is given in Figure 4.3-1 and Figure 4.3-2.

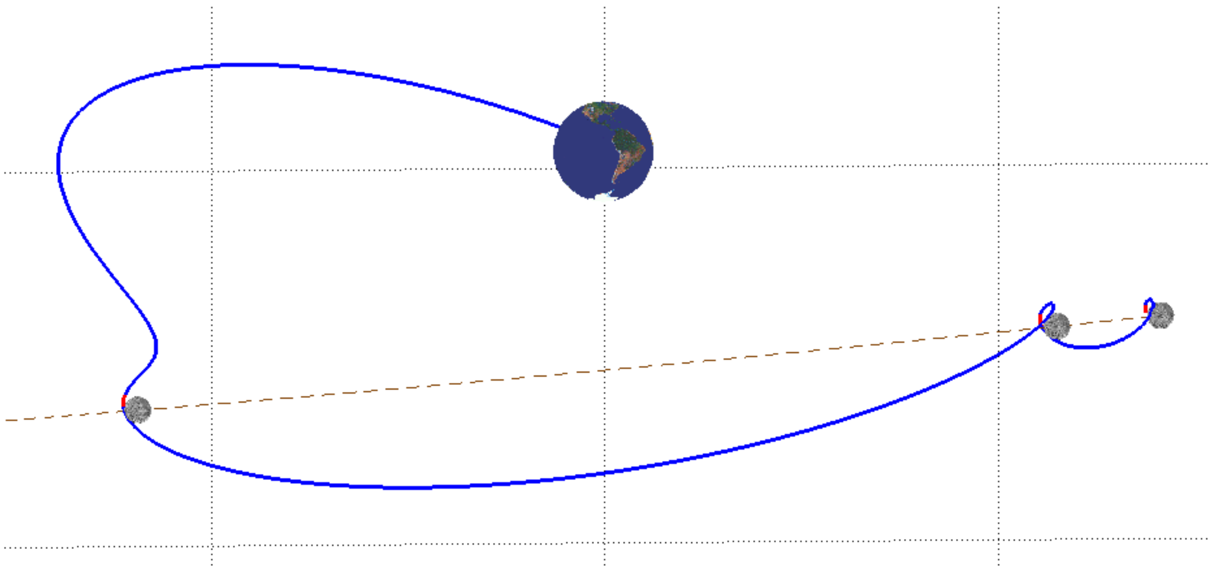


Figure 4.3-1: Optimized trajectory including lunar orbit injection plotted in ECI system. The position of the Moon is plotted at the beginning of each thrust arc (red).

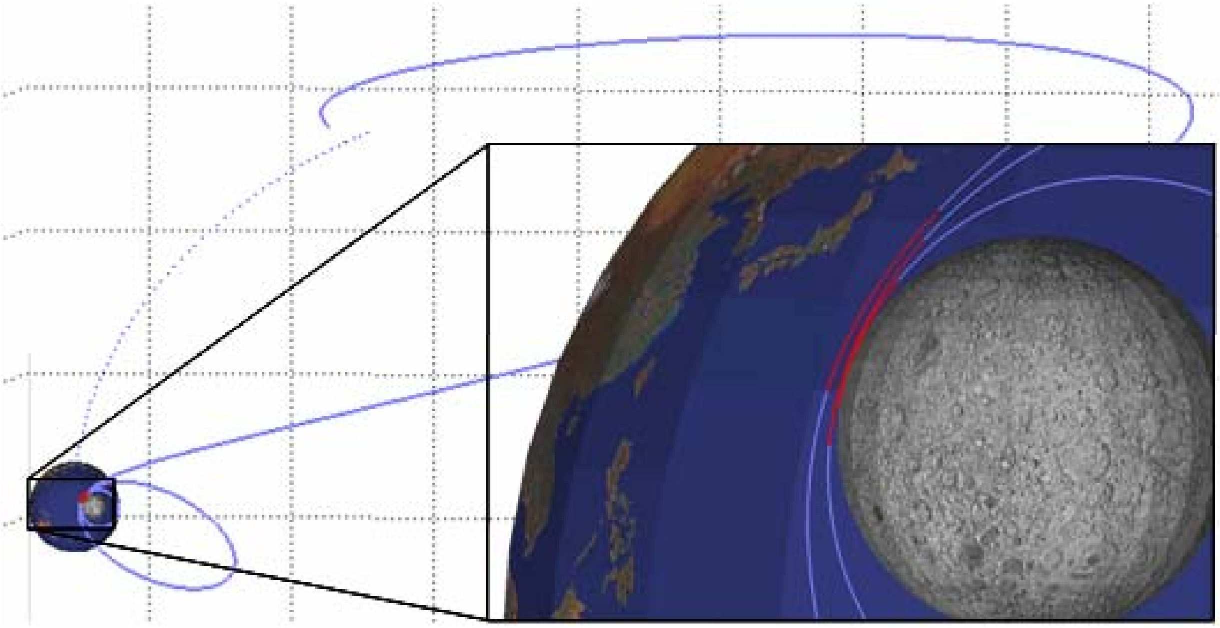


Figure 4.3-2: LTO and LOI in the moon fixed coordinate frame. The dotted line shows the Earth orbit during the mission. The magnification on the right side shows that the thrust arcs will be visible from Earth. The rotation of Earth and Moon are not considered within this graphical representation.

The injection into lunar orbit depends on the characteristics of the propulsion system and the resulting limitations (e.g. thermal load, specific impulse, thrust, and maximum burn time). Furthermore, it has to be taken into account that gravitational losses increase with longer burn durations at each LOI manoeuvre, whereas the risk of being recaptured by Earth's gravity rises with shorter burn durations. The detailed analysis was performed w.r.t. to these obvious limitations. Possible additional constraints from navigation and instrument requirements have not yet been taken into account.

The target orbit for the initial commissioning is a 100 km circular Moon orbit with 85° inclination. In addition to the nominal manoeuvre, a sensitivity analysis has been performed, taking into account an underperformance of the first injection manoeuvre. Figure 4.3-3 illustrates the orbits achieved after LOI-1, considering different levels of under-performance. Simulations also show that for an under-performance of 30% (70% of the nominal ΔV is applied), the spacecraft ends in an Earth orbit.

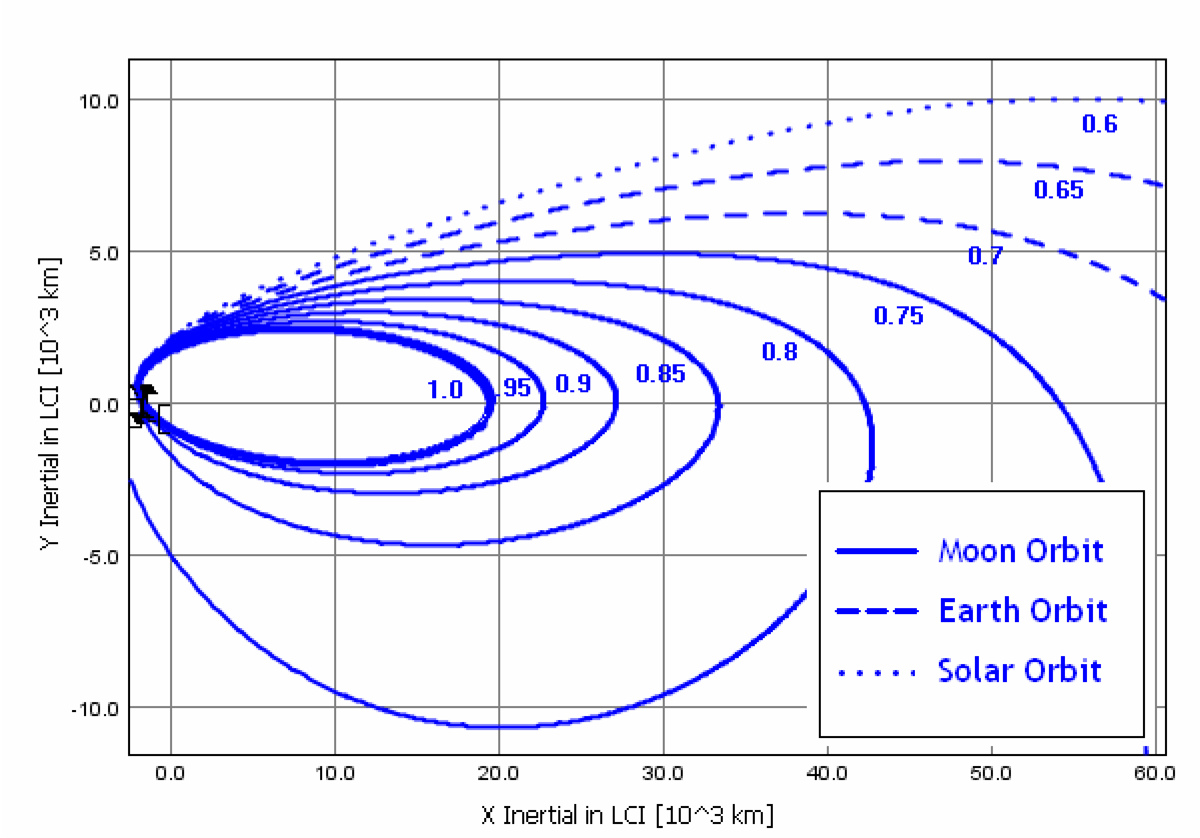


Figure 4.3-3: Trajectories after the first LOI with the performance of LOI-1 varying from 60% to 100%

4.3.2 Lunar Science Orbit

Once the spacecraft is placed into the 100 km circular commissioning orbit and commissioning is finished, the orbit altitude is lowered to the 50 km operational orbit. Depending on the experience gained during the commissioning phase, this maneuver can be done in one or several steps, ensuring that the final lunar science orbit is reached with a very high accuracy. A detailed analysis of the lunar science orbit is presented in [RD18].

One basic requirement for the mission is to ensure a sufficient electrical power supply. It turns out, that in contrast to the Earth, it is not possible to achieve a sun-synchronous orbit around the moon because the nodal drift is too small (small J2 term in the Moon's gravity field). For polar moon orbits, the inertial orientation of the orbit plane almost remains constant throughout the year. I.e., the sun will rotate around the orbital plane once per year. In order to ensure electrical power supply and a good thermal environment, this will require a yaw manoeuvre of 180° every 6 months. The principle geometry is illustrated in Figure 4.3-4.

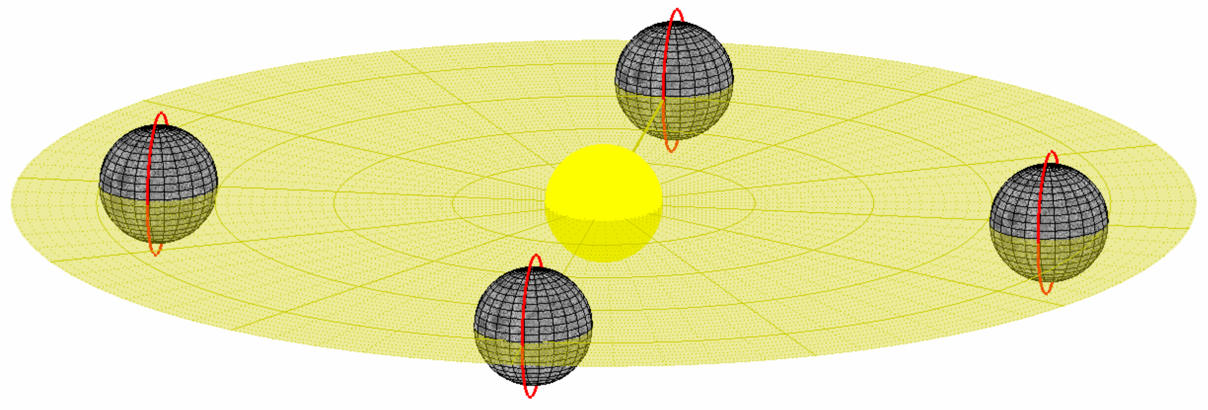


Figure 4.3-4: Principle sketch of the orbit geometry w.r.t. the sun throughout one year (not to scale).

Besides the general geometry, eclipse times must be taken into account. The normal night duration when the orbiter is passing through the moon shadow have a duration of 43 to 45 min. During lunar eclipse times, the shadow from the Earth must be taken into account as well and for such special constellations, eclipse times with no or significantly reduced sun-light can increase up to approx. 200 min plus 125 min of reduced power while the moon is still in the half-shadow of the Earth. The exact duration of such events must be analyzed once the launch date is known. For the LEO phase A analysis the lunar eclipse of 27 July 2018 was considered as the worst case.

From an operational point of view, it must be ensured that the ground contact times are sufficiently frequent and long such that all relevant data can be transferred to ground and a sufficient health monitoring and telecommanding can be performed. Over a month, there will be a variation in the visibility of the moon orbit as illustrated in Figure 4.3-5.

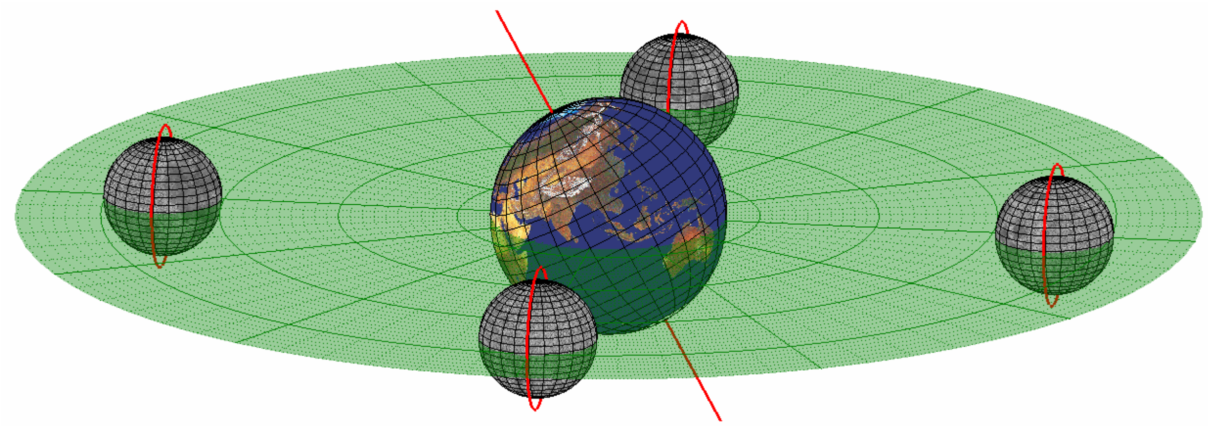


Figure 4.3-5: Visibility of the operational orbit around the moon as seen from Earth (not to scale).

The most important scientific requirement of LEO is a global coverage of the moon surface with all instruments. In this context, the specific requirements from the different instruments must be taken into account. It has already been shown in the context of the phase 0 study, that for a 50 km reference altitude, natural altitude variations of ±25 km will occur throughout the months due to the irregularities in the Moon's gravity field. Already in phase 0, it has been found that around 85° inclination, orbits with a natural lifetime of several years exist. On the other hand, polar orbits with 90° inclination require approximately 150 m/s/year of ΔV for orbit maintenance in order to avoid a deformation of the orbit that causes a crash of the spacecraft into the Moon's surface. This analysis was significantly extended in the context of the phase A study in order to improve the understanding of the underlying phenomena and perform a more refined definition of a reference orbit.

4.3.2.1 ΔV-budget

From the result of the mission analysis a ΔV-budget has been established that takes into account all orbit manoeuvres of the LEO mission. The following margin have been applied:

- 5% ΔV margin on manoeuvres that have been calculated in detail, i.e. including gravity losses
- 10% ΔV margin fro manoeuvres which have been calculated without gravity losses
- 100% ΔV margin for manoeuvres which depend on still uncertain environmental parameters

Table 4.3-1 LEO ΔV-budget

Manoeuvres in Lunar Orbit	Nominal ΔV [m/s]	Margin [%]	ΔV [m/s]	Isp [s]	Remark
Orbit lowering from 100 km to 50 km	23	10%	25	272	Hohmann transfer
Excentricity control in 85° Orbit	36	100%	72	272	for 3 years
Inclination control in 85° Orbit	25	100%	50	272	for 3 years
Inclination change from 85° to 90°	150	10%	165	272	1 time
Orbit mainatinance, polar Orbit	300	10%	330	272	for 1 year
Total, Main Satellite only	**511**	20.7%	**617**	272	
Total of stack (main + subsatellites)	**23**	10%	**25**	272	

Manoeuvres during Transfer	ΔV [m/s]			Isp [s]	Remark
Transfer-orbit correction	75	100%	150	319	Launcher dispersion correction and targetting
Lunar orbit insertion	780	5%	819	319	according to ASTOS detailed calculation
Total	**855**	13.3%	**969**		

4.4 Mission Operations

This chapter focuses on the mission phases of the LEO mission and gives an overview about the main events that occur during each mission phase. The concept for the operations of the LEO satellites by the ground segment is described in Chapter 4.6 below. Detail on the spacecraft and instrument operations can be found in [RD20].

The LEO mission phases are defined for the Main Satellite and for the Sub-satellites. Up-to and including Pre-Launch Phase the mission phases are common for the main satellites and the Sub-satellites. For the subsequent parts of the mission, the mission phases for the Main Satellite and the Sub-satellites differ.

The LEO Main Satellite mission is defined into 14 separate mission phases. The Sub-satellites mission is defined into 10 separate mission phase. During six of these, the Sub-satellites are still attached to the LEO Main Satellite. Table 4.4-1 shows the individual mission phases with start/end triggers and their major objectives for the LEO Main Satellite. Table 4.4-2 does the same for the Sub-satellites. The mission phases will be referred to in the following chapters by the identifiers that are also listed in the tables. The Main Satellite phase identifiers are prepended by “L-”, the Sub-satellite phase identifiers are prepended by "S-". Common mission phases do not have a prefix.

Table 4.4-1 Mission phases of the LEO Main Satellite

Phase	Identifier	Begin	End	Duration	Objectives
Manufacture, Assembly, Integration & Test	AT	Successful CDR	SC leaves industrial contractors site	TBD	Manufacture, Assembly, Integration and Testing of the Main Satellite
Transportation	TR	SC leaves industrial contractors site	Arrival at launch site	TBD	Transport of LEO Space Segment to launch site
Conditioning and Tests	CT	Arrival at launch site	Begin of installation on launcher	TBD	Fuelling and test of LEO
Installation on Launcher	IN	Begin of installation on launcher	Completion of installation on launcher	TBD	Mounting of LEO Space Segment on Launcher
Pre-Launch	L-PL	Start of LV Countdown Sequence	LV Lift-off	~1 day	Configure LEO into Launch Mode Short LEO Checkout
Launch	L-LA	LV Lift-off	Launcher Separation	~90 min	Achieve Trans-Lunar Trajectory
Early Cruise	L-EC	Launcher Separation	Normal Mode	~ 6 h	Sun Acquisition and Ground Acquisition Initial MCC Planning
Mid Cruise	L-MC	Normal Mode	Completion of MCC	~1 day	Propulsion Checks Final MCC Planning Execution of MCC Burn
Late Cruise	L-LC	Completion of MCC	Start of LOI Sequence	~3-4 days	LEVIS & RadMo Turn-On Activities (TBC) Spacecraft Functional Checkout LOI Planning
Lunar Orbit Acquisition	L-LI	Start of LOI Sequence	Commissioning Orbit	~4-6 days	Perform Lunar Capture Manoeuvre Achieve 100x100 km, 85 deg incl. Commissioning Orbit
Commissioning	L-CO	Commissioning Orbit	85 deg Mission Orbit	~60 days	Deployments: Solar Array, HGA Spacecraft Checkout and Calibrations Mission Orbit Adjustment Sub-satellite Deployment

Phase	Identifier	Begin	End	Duration	Objectives
Nominal Mission 1	L-N1	85 deg Mission Orbit	90 deg Mission Orbit	~3 years	Routine Operations Non-Routine Operations Data Product Generation
Nominal Mission 2	L-N2	90 deg Mission Orbit	1 Year in Polar Orbit	~1 year	Routine Operations Non-Routine Operations Data Product Generation
Extended Mission	L-EX	1 Year in Polar Orbit	Impact	TBD	Goals to be Determined Impact Prediction/Activities
Disposal	L-DI	Impact	Completion of Close-Out Activities	N/A	Finalise Mission Operations/Activities

Table 4.4-2 Mission phases of the Sub-satellites

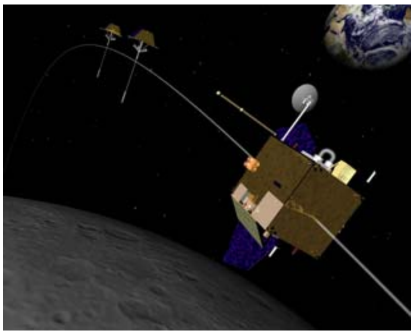
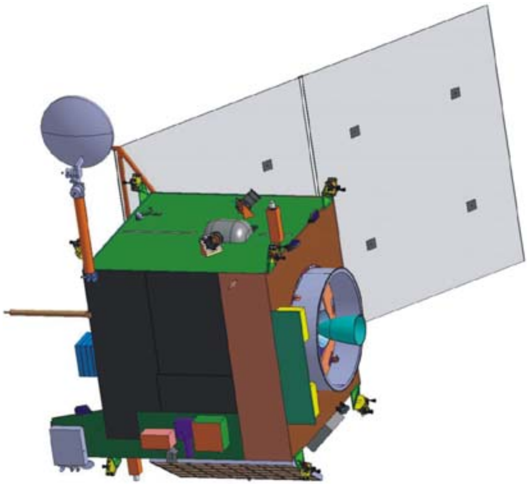
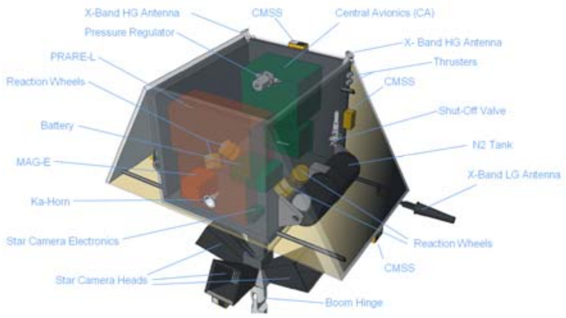
Phase	Identifier	Begin	End	Duration	Objectives
Manufacture, Assembly, Integration & Test	AT	Successful CDR	SC leaves industrial contractors site	TBD	Assembly, Integration and Testing of the Sub-satellites
Transportation	TR	SC leave industrial contractors site	Arrival at launch site	TBD	Transport of LEO Space Segment to launch site
Conditioning and Tests	CT	Arrival at launch site	Begin of installation on launcher	TBD	Fuelling and test of Sub-satellites
Installation on Launcher	IN	Begin of installation on launcher	Completion of installation on launcher	TBD	Mounting of LEO Space Segment on Launcher
Pre-Launch	S-PL	Star of LV Countdown Sequence	LV Lift-off	~1 day	Configure Sub-satellites into Pre-Launch Mode Short Sub-satellites Checkout Configure Sub-satellites into Off-Mode
Transfer	S-CR	LV Lift-off	Sub-satellites Power ON	~ 66 days	N/A
Checkout	S-CH	Sub-satellites Power ON	Deployment	~7 days	Sub-satellites Checkout and Calibrations (TBC) Deployment of LunarMag Acquisition of Deployment Orbit by LEO Sub-satellites Deployment
Commissioning	S-CO	Deployment	Nominal Orbit	~20 days	Sub-satellites Checkout and Calibrations Sub-satellites Formation Acquisition
Nominal Mission	S-NM	Nominal Orbit	TBD years in Nominal Orbit	4 years (goal)	Routine Operations Non-Routine Operations Data Product Generation
Extended Mission	S-EX	TBD Years in Nominal Orbit	Impact	TBD	Goals to be Determined Impact Prediction/Activities
Disposal	S-DI	Impact	Completion of Close-Out Activities	N/A	Finalise Mission Operations/Activities

4.5 Space Segment

4.5.1 System Overview

The main characteristics of the space segment are summarised in Table 4.5.1. The design of the space segment is still evolving and hence the characteristics will be subject to change that will be reflected in later versions of this document.

Table 4.5-1 Main characteristics of the LEO Space Segment

Space Segment Elements • 1 Main Satellite • 2 identical Sub-satellites • Maximal total launch mass 2050 kg	
Main Satellite • Dimensions: 2000 x 1900 x 2000 mm³ • Dry-mass ca. 850 kg • Propellant mass ca. 800 kg • Onboard propulsion system for lunar orbit insertion and orbit and attitude control • Solar array with 1 degree of freedom • HGA with 2 degrees of freedom • TT&C via X-band • High attitude control performance • Payload data transmission via Ka-band	
Sub-satellites • 2 identical Satellites • Dimensions: 800 x 1400 x 680 mm • Dry mass ca. 120 kg • Propellant mass ca. 7 kg • Cold-gas propulsion system for orbit and attitude control • Minimal number of moving parts • Stable centre of mass • High magnetic cleanliness	

4.5.2 Main Satellite

4.5.2.1 Main Satellite Concept

The corner stones of the main satellite design have already been established in the assessment phase of the LEO project. The changes established during the phase A mainly reflect the goal to use the heritage of the Small-GEO platform for the main satellite and to facilitate AIV in later phases. Here we briefly summarize the main design features of the LEO main satellite.

Instrument accommodation

The preferred observation direction of most instruments is nadir. In addition some instruments (e.g. SERTIS) require a deep space view for calibration. Together with the form factor of the various instruments this has lead to a coherent approach on the instrument accommodation:

- Narrow angle optical instrument are place on a side panel that is kept in permanent shadow from the Sun facing nadir.
- All other instrument that need a nadir view are placed on the nadir panel

In order to ensure that the instruments are in permanent shadow from the Sun, the main satellite conducts a flip around its yaw axis every 6 months.

Power supply

A single one degree-of-freedom solar array is used in order to cope with the varying illumination conditions at the Moon. The solar array is mounted on one of the side panels of the platform at an angle of 30° with respect to the panel. This configuration optimises the illumination of the Solar array in combination with the above mentioned yaw flip. Even with a one degree-of-freedom solar array the Sun incident angle on the solar array varies significantly. Hence the solar array is connected via a maximum power point tracker to minimise conversion losses

Telecommunications

A dual band system has been selected for telecommunications. This is in part dictated by ITU regulations and partially driven by the high payload data rate. For LEOP a communication system in S- or X-band is required in order to be compatible with the current LEOP networks. X-band was selected for LEO because they are no S-band frequencies available anymore for use at the Moon due to the large number of other planned Moon missions and hence only an X-band system can still be used for TT&C in lunar orbit. Even in X-band the available bandwidth is restricted to 50 MHz by the ITU and hence an X-band link is insufficient for the payload data transmission. Hence an additional downlink in Ka-band is foreseen for LEO. During nominal operations at the Moon the links are via an HGA, during transfer and in safe mode only the X-band link is established via two LGAs with hemispherical coverage. The HGA has two degrees of freedom and is mounted on a boom to improve visibility of the Earth. The HGA was chosen to have a diameter of 0.7 m which turned out to be a good compromise between antenna gain and minimising the disturbances onto the spacecraft due to the motion of the antenna.

Propulsion

A bi-propellant propulsion system has been selected for the LEO main satellite in order to maximise the specific impulse and hence minimise the propellant consumption. This is necessary due to the considerable ΔV demand, both during lunar orbit insertion and for orbit changes and orbit maintenance in lunar orbit.

Thermal control

Thermal control in the lunar environment was found a particular challenge. In its low orbit around the moon LEO receives a considerable amount of infrared radiation from the lunar surface during day and nearly none during night. In order to cope with challenging thermal cycle, the shadow panel which hosts most of the radiators is shielded form lunar infrared radiation by a grating of shades. The potential accommodation of radiators on the Zenith panel, which is not expose to lunar radiation, has been discarded because most heat dissipating equipment has to be accommodated rather far from it.

Payload to Platform Interface´

The large number of 12 instruments on the LEO main satellite requires special planning for AIT and operations. In order to improve these aspects a special payload support system (PSS) was introduced that centralises the interfaces between the platform and the instruments. The PSS consists out of a payload power distribution unit, a payload management computer, a payload mass memory and the complete payload data transmission chain. The PSS interfaces with the platform only via a redundant Milbus line, a 28 V regulated power supply and some dedicated lines. In this way the payloads can be pre-integrated and tested together with the PSS with minimal impact on the platform AIT. Not only this largely decouples the platform and instrument schedules but it also makes the task of instrument significantly easier to handle. Also the operations of the instruments during the mission will be handled by the PMC with minimal action of the platform OBC.

Attitude control

The optical payloads of LEO require a high performance attitude control system with both high absolute pointing accuracy and high pointing stability. The layout of the AOCS system has not yet been finalised and more options need to be considered. However it has been shown that an option using startrackers and an IMU fulfil the requirements. Further analysis of the AOCS system will also have to consider the effect of thermal deformations of the satellite structure which have not yet been taken into account.

Structure

The platform responsible OHB System has identified the SGEO platform as a suitable starting point for the LEO main satellite. The SGEO platform already provides a suitable bi-propellant propulsion system and can accommodate a payload of up to 300 kg on its front panel which is more than sufficient to carry the subsatellites. The structure of SGEO is based on internal crossed shear walls. The tanks have polar mountings, with the lower mountings being attached to the launcher adaptor and the upper mountings on a cross shear walls. For LEO the propellant tanks can be scaled down. Hence the structural loads will be reduced compared to SGEO.

The main satellite is described in more detail in the following chapters.

4.5.2.2 Platform

According to the System Functional Specification [AD2] and the GDIR [AD4], a baseline concept for the main satellite platform was elaborated. This baseline concept is described in the following section.

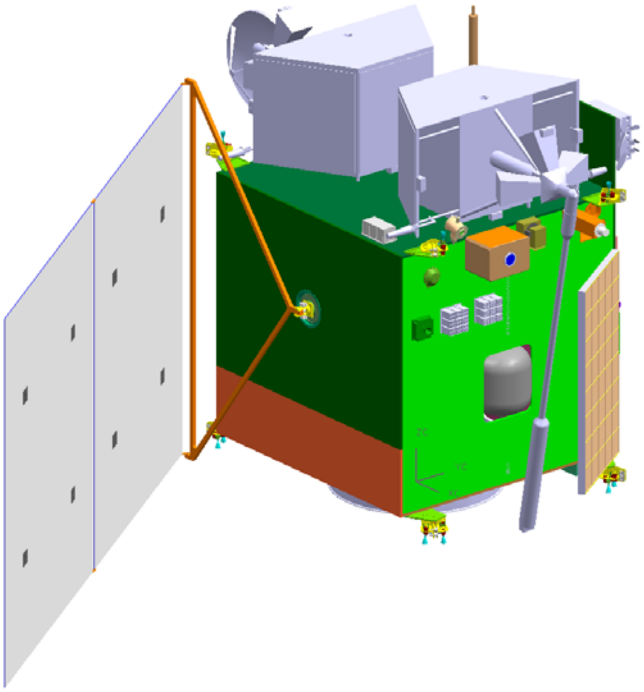


Figure 4.5-1 LEO external geometry

The satellite platform contains all subsystems which are required for the operation of the satellite. A detailed accommodation study of the platform was performed and summarized in the following figures. The platform accommodation is as far as possible identical to the PDR SGEO accommodation.

The subsystems are characterized as following:

- OBDH Subsystem

  The OBDH System controls the single subsystems by transferring and processing commands and acquiring sensor information and other data.

- OBDH Software

  The data processing compiles the data about the satellite status and position and submits the data during contact to the satellite control. Thereby it receives commands for manoeuvring profiles, system parameters, and reloadable software which controls the subsystems. The data processing subsystem handles also telemetry data storage and the transmission to the ground station.

- TT&R Subsystems

  Telemetry, Telecommand and Ranging Datalink: TM/TC Transceiver Unit receives commands from the satellite control ground station and transmits satellite status data to the ground station. The orbital position is acquired via ranging from ground.

- Attitude & Orbit Control System

- ACS: The function is to control the standard attitude, to perform the reorientation manoeuvres for nominal functions and emergency attitude control, in which the satellite will be manoeuvred in a stable attitude to maintain the vital functions. The ACS features stabilization via momentum wheels and attitude information via star sensors and a gyro unit. For contingency operation coarse sun sensors are added to acquire attitude data.
- OCS: It is used to provide the required functions in order to perform the Moon obit injection, maintain the satellite in its orbit, and perform the inclination change. The orbital position is acquired via ranging from ground.

- Propulsion Subsystem

  The actuator system consists of the bipropellant, MMH/MON propulsion system (PPS) in mini-blow down mode. Though the PPS is an actor for the AOCS, this subassembly is handled as a separate subsystem due to its complexity

- Structure

  The structure contains all subsystems of the satellite bus and assimilates the launch loads. The structure consists of separate modules for Platform and Payload.

- Thermal Subsystem

  The thermal subsystem provides all functions for maintaining all component temperatures within the tolerable limits. It provides e.g. cooling of the electrical components and heating of PPS components.

- Electrical Power Subsystem

  The power will be generated by the Solar Panel, supplies components by a distribution system and is transferred to batteries for the eclipse mode. The LEO power system consists of dedicated solar arrays, batteries and power supply electronics. It supplies the LEO satellite platform and the payload power distribution.

Modular Design

The design of the LEO satellite follows strictly a modular approach. The spacecraft is divided into the major modules (see Figure 4.5-2)

- Core Platform Module
- Propulsion Module

forming the Platform, and

- Payload Module
- Subsatellites

forming the Payload of the LEO mission.

This modular approach enables a parallel AIV process, reducing cost and supporting a fast realisation time.

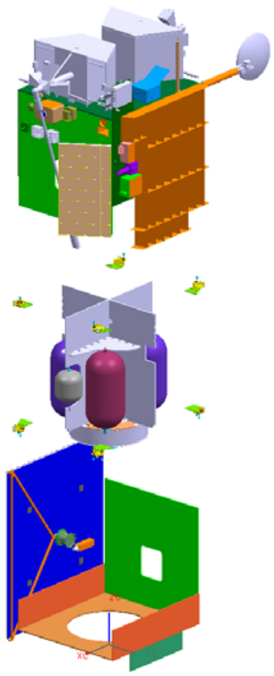
Payload	**2 Subsatellites**	
	Payload Module	
Platform	**Propulsion Module**	
	Core Platform Module	

Figure 4.5-2 LEO satellite elements

Platform Key Figures

Table 4.5-2 shows a summary of the key features of the Phase A baseline LEO platform.

Table 4.5-2: LEO Platform Key Features

General	
Launcher Compatibility	Soyuz, Zenith (Land Launch) and Falcon 9 (TBC)
Satellite Lifetime	> 4 years in Moon orbit environment
Platform Dry Mass (incl. margins)	< 605 kg
Satellite Body Dimensions	2000 mm x 1900 mm x 2000 mm
Payload Envelope	
Application	Science
Max., effective P/L Power	< 800 W
Total Payload Mass	< 250 kg
Main Payload Bus Voltage	regulated 28.0 V_{dc}
Telemetry, Telecommand & Ranging (TT&R)	
Frequency (receive/transmit)	X- Band
TT&R Antennas	LEOP, Contingency: Near-omnidirectional antennas Operational: High gain communications antenna (2DoF)
Propulsion Subsystem	
On-Station	12 x 10N, bipropellant MMH/MON in Mini-Blow-Down mode
Moon Transfer	1x 400N main thrust, bipropellant MMH/MON in pressure regulated mode
Electrical Power Subsystem	
Av. Power Generation Capacity	< 1500 W EOL
Eclipse Capability	battery max 26% EoL DoD
Main Platform Bus Voltage	Unregulated 50.0 V_{dc}
Energy Storage	Li-Ion Technology
Solar Array	12.1 m^2, Triple-junction GaAs solar cells
Attitude Control Subsystem	
Stabilization	3 axis stabilized
Pointing Accuracy	0.1°
Pointing Knowledge	5“ (or 24.2407 µrad)
Pointing Stability:	0.5“ / 0.56s (or 2.42470 µrad / 0.56s)
Command & Data Handling	
Central Platform Data Bus	MIL Standard 1553B
Flight Processor	LEON II - FT

SGEO Adaptation

The requirement of LEO main satellite platform concerning propulsion, payload mass and structural loads are similar are similar to those of the SGEO platform. Hence an adaptation of the SGEO platform for use as the LEO main satellite is foreseen. The principle behind the reuse of the SGEO platform for LEO is shown in Figure 4.5-3.

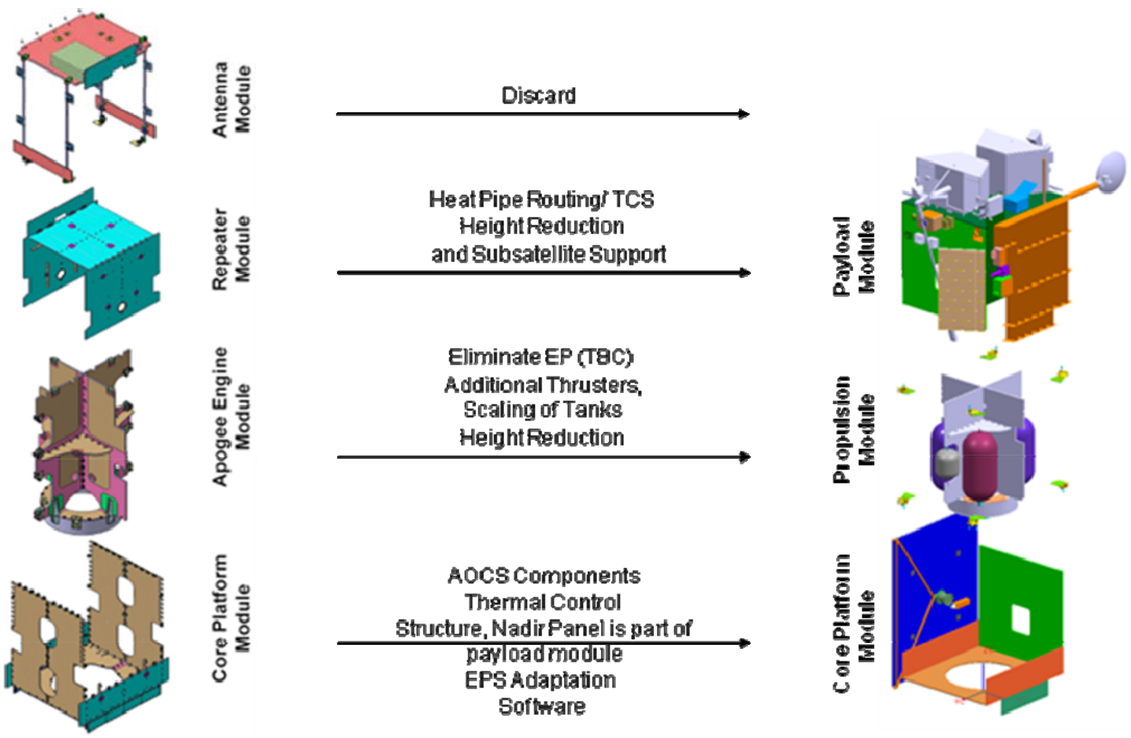


Figure 4.5-3 SGEO Adaptation

Platform Accommodation

Figure 4.5-4 shows the external platform components accommodation. The single subsystem studies, as well as the accommodation study, show that an absolute separation of payload and platform module is not feasible. On the payload module following platform components have to be accommodated:

- SADA
- Sun Presence Sensors
- Thrusters
- TT&R Omnidirectional antennas

The AIT process must foresee the early as well as late project actions to implement these components and its interfaces and support structures (pipes, harness, TM sensors).

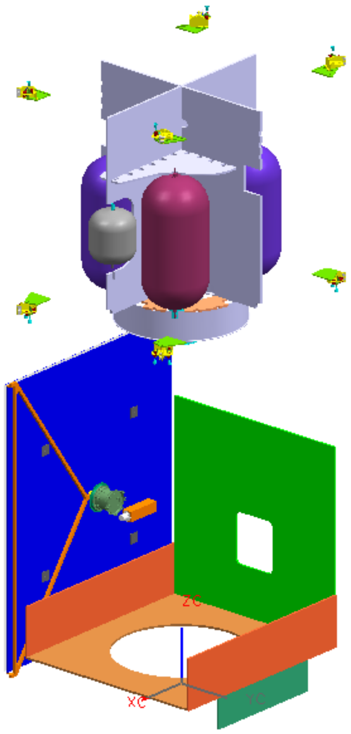


Figure 4.5-4 Bus Components Accommodation

Figure 4.5-5 and Figure 4.5-6 show the accommodation of the platform components. The accommodation of the dissipating units was mainly done on the radiator panels. The density of the components was chosen with respect to harness/ connector and thermal aspects. Nevertheless the other panels are offering additional area and margin for potential, additional hardware.

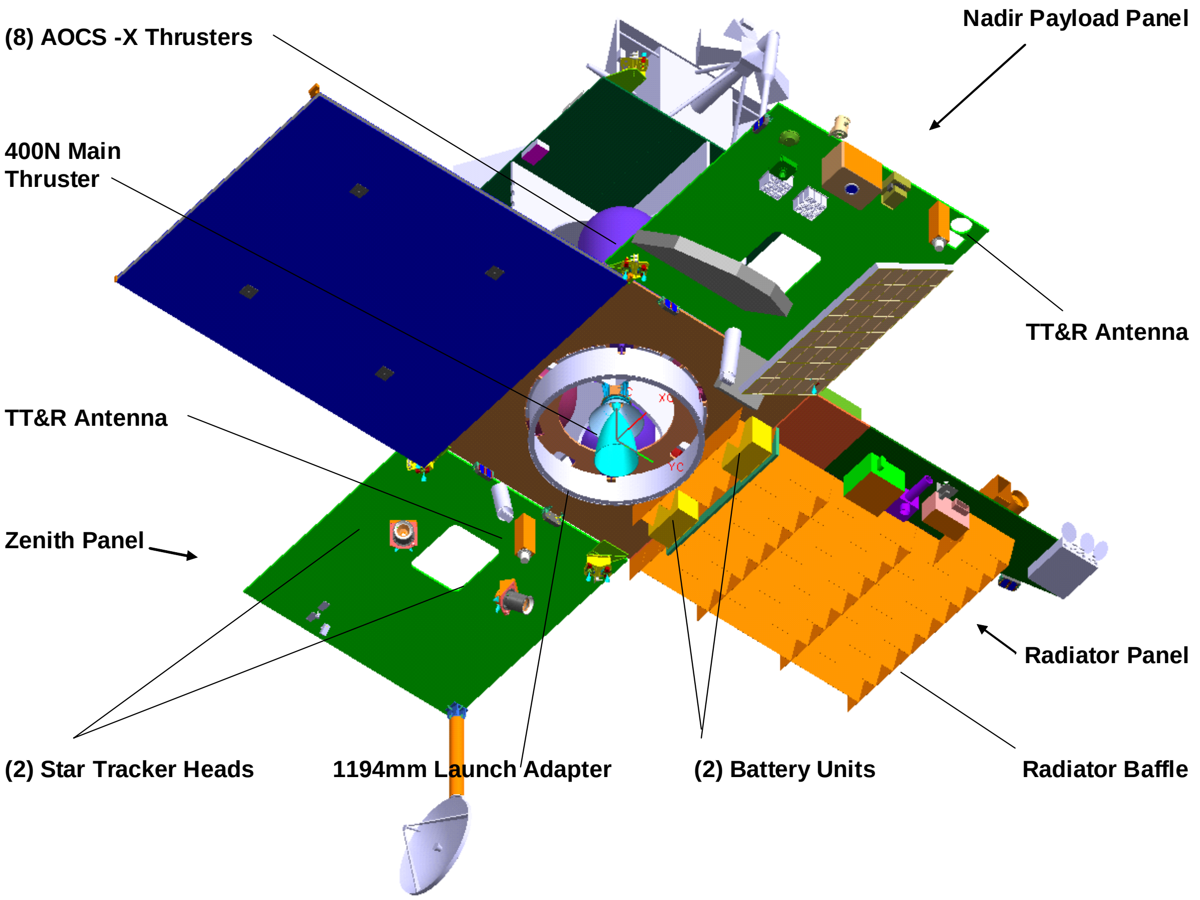


Figure 4.5-5 Internal Bus Components Accommodation (1)

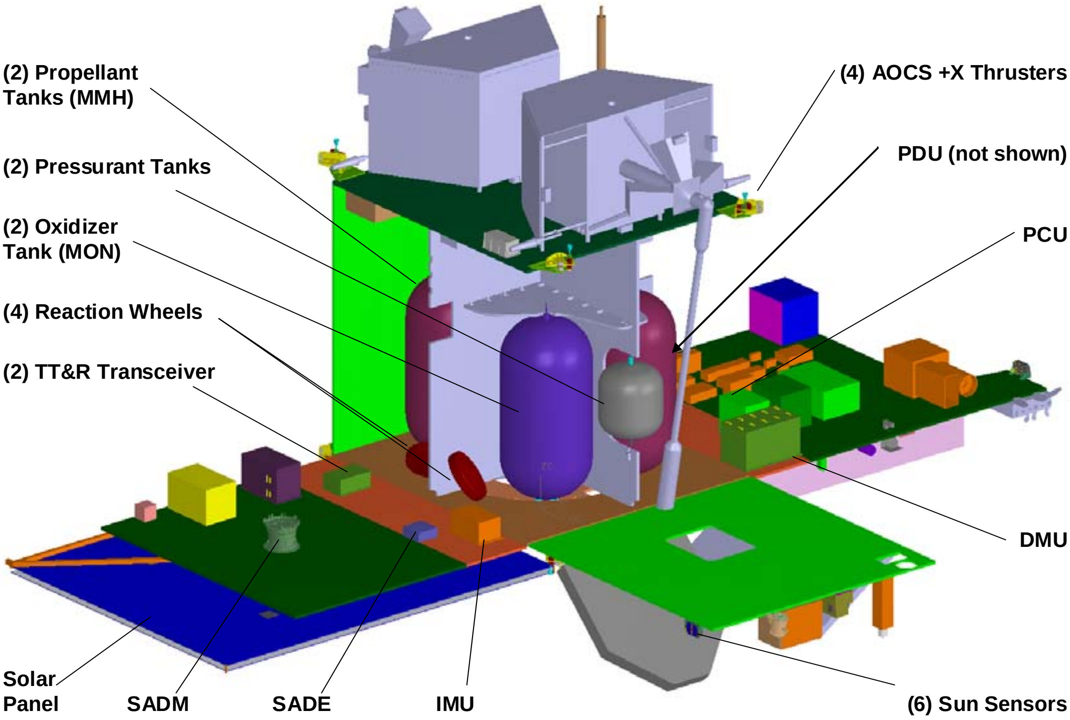


Figure 4.5-6 Internal Bus Components Accommodation (2)

<u>Electrical Power Subsystem</u>

Table 4.5-3 shows a short requirement analysis for the EPS.

Table 4.5-3 Requirement analysis for LEO EPS

Requirement	met?	Remarks
General		
Redundancy	yes	by design: SA and battery divided in substrings; all PCU components at least twice
Supply electrical power in eclipse	yes	
Supply of P/L with 28V regulated	yes	
Supply of platform with 50V unregulated	yes	
Transfer orbit		
Supply bus with power	yes	as SA is deployed directly after launch and S/C can be aligned to sun, max. el. power is available
Operational orbit at max. eclipse		
Gather enough energy for sun and eclipse phases	yes	
Operational orbit pure sun		
Gather enough energy for pure sun orbit	yes	
Lunar eclipse*)		**see below**
keep defined systems supplied	yes	battery can once be discharged >26% DOD
Abnormal orbits)**	**open**	**TBD, see below**

***)Lunar eclipse:**
During this eclipse the orbiter stays in its worst case for 2h 15min without sunlight. According to the power budget in section 4.5.2.4, the power demand in this phase is 454W (energy demand 1022Wh). This discharges the battery (C = 3110Wh) to a DOD of 33%. This value is acceptable, as it is only slightly higher than the normal operational DOD and the lunar eclipse happens only once during the mission.

****)Abnormal orbits:**
The EPS design assumes nominal orbits of maximum 48min eclipse and minimum 65min sun phase. If orbits occur, that deviate for a longer time significantly of these values, this may have to be considered in a new design.

The electrical power system of LEO is shortly characterized as follows:

- Unregulated bus of 50V for the platform (33.6…50.4V)
- Regulated bus of 28V for the payload (28 +/-1V)
- Bus power about 1kw

Main components are:

- Solar array (SA)
- Solar array drive mechanism (SADM)
- Battery
- PCU (with SA regulator, 50 to 28V conversion for P/L)
- PDU

The following picture shows the EPS block diagram.

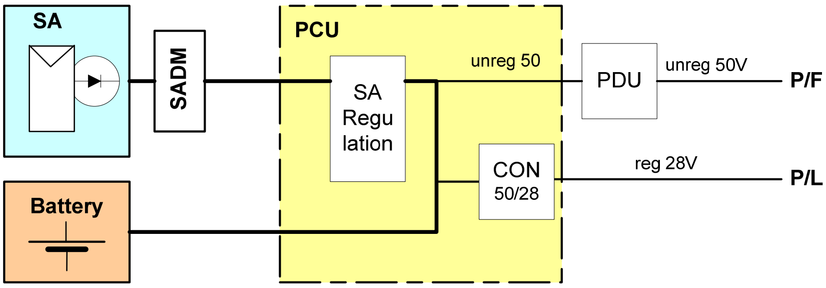


Figure 4.5-7 Principal block diagram of LEO EPS

Solar Array Regulation

A trade comparing the mass, efficiency and heritage of two regulation principles was performed (see [RD21]) between

- S3R (series shunt switching regulation) and
- MPR (maximum power point regulation)

The MPR regulation principle was chosen, with the main arguments being:

- MPR enables more output power at the limited SA area
- with MPR, SA panels of SGEO can be used in their original state

MPR is roughly characterized by:

- SA load impedance controlled
- Energy transfer via converters
- Efficiency of regulation unit itself less than S3R
- Regulation unit more complicated
- SA operation in maximum power point
- No SA mismatching, high overall efficiency of EPS

Solar Array (SA)

The SA is the only energy source for the orbiter during its mission. The LEO SA is mechanically a derivate of the SGEO panels. It is limited to two panels of 2.216 x 2.73 m^2 each. Electrically only the MPR design can use the SGEO panels in their original state. The cell technology is high efficiency GaAs. The electrical SA design for LEO is performed in principle as follows:

The worst case for the EPS is an orbit with maximum eclipse. So for the SA design this orbit must be the starting point. For this orbit, it is necessary:

- Identify energy demand for sun and night phase without PCU losses but incl. margins
- Calculate energy demand for sun phase including battery charging under consideration of battery efficiency
- Calculate from this the average power demand in sun phase (including battery charging)
- Calculate from this under consideration of EOL, temperature, sigma, SA matching, PCU efficiency and cell datasheet the nominal SA power (BOL, 28°C)
- Configure SA layout (cells in series and parallel strings) by means of the cell data sheet
- Verify that this SA (with the chosen sigma) is also suitable for pure sun orbits
- Verify that this SA is also suitable for lunar eclipse and possible further special situations

The SA design does not include its own power margins as they are defined already in the power budget. Another margin refers to the maximum SA temperature. This is calculated as 115°C during orbits with maximum eclipse and as +60°C during pure sun orbits; SA design adds 20K to these values.

The following constants have been identified for the LEO SA design:

Table 4.5-4 Constants for LEO SA design

Cell type		Azur 3G-28% (GaAs/Ge)	
Degradation for EOL		2.5E14	
eta(Battery)		0.95	efficiency of Li-Ion battery
eta(PCU)		0.87	MPR, 50V, unreg (incl. 50/28V conv.)
matching (SA)		0.98	MPR, all conditions
eta(SA) EOL,temp		0.87	efficiency of real SA at EOL and SA-temp –120..+115°C compared to nominal SA
temp(SA)	°C	-120…+115 +60	max. eclipse orbit pure sun orbit
sigma	°	60	fixed SA angle
Duration max eclipse	min	48	
Duration orbit	min	113	

SADM

The solar array drive mechanism enables the orbiter to rotate its solar array around one axis. The power of the SA is transferred via several slip rings to the bus. The SGEO SADM design can be applied also for the LEO mission..

Battery

The battery is of Li-Ion technology. Data of the battery cell ABSL 18650HC were used for reference.

Table 4.5-5 LEO battery data

LEO Battery Data			
	No.	**Description**	
Battery cell		ABSL 18650HC	
cycles		20000	
E11	Wh	813.6	energy demand during eclipse
Capacity	Wh Ah	3110 72	BOL
DOD	%	26	BOL
Layout		12s48p	12 cells in series, 48 parallel
Voltage	V	33.6…50.4	per cell: 2.8V…4.2V
I/F temperature	°C	+5…+15	
Mass total	kg	31.9	
Dimensions (2 modules, each)	mm^3	262x269x170	modules of TerraSar-X

The a.m. battery design has already be flown in the TerraSar-X project (with three modules).

PCU and PDU

It is not clear by now, whether for LEO the PCU and PDU are combined in one box (as shown in Figure 4.5-8). The actual SGEO baseline foresees the distribution of these tasks to two separate boxes, especially due to testability and verification approach. Driven by cost and reliability considerations this concept is baselined on system level.

An alternative concept is the combination of both functions in one box, reducing the number of boards e.g interface to Milbus 1553. For LEO, it shall be analysed specially with focus on the different test approaches. Following paragraphs are showing a possible design concept, which is also based on flight proven hardware, but not based on the SGEO baseline.

The PCU and PDU will communicate to the platform via Milbus 1553. Additionally they are equipped with discrete lines for emergency warnings in case of undervoltage (TBC).

The PDU part only distributes power to platform subsystems. Power distribution for payload subsystems is performed in the PPDU, which does not belong to the EPS.

The existence of an additional unregulated 50V line to the payload in addition to the 28V line is currently not in the baseline. Such a line could however serve for P/L heater and other P/L equipment, that do not require reg. 28 V. Capability and protection of this line is TBD.

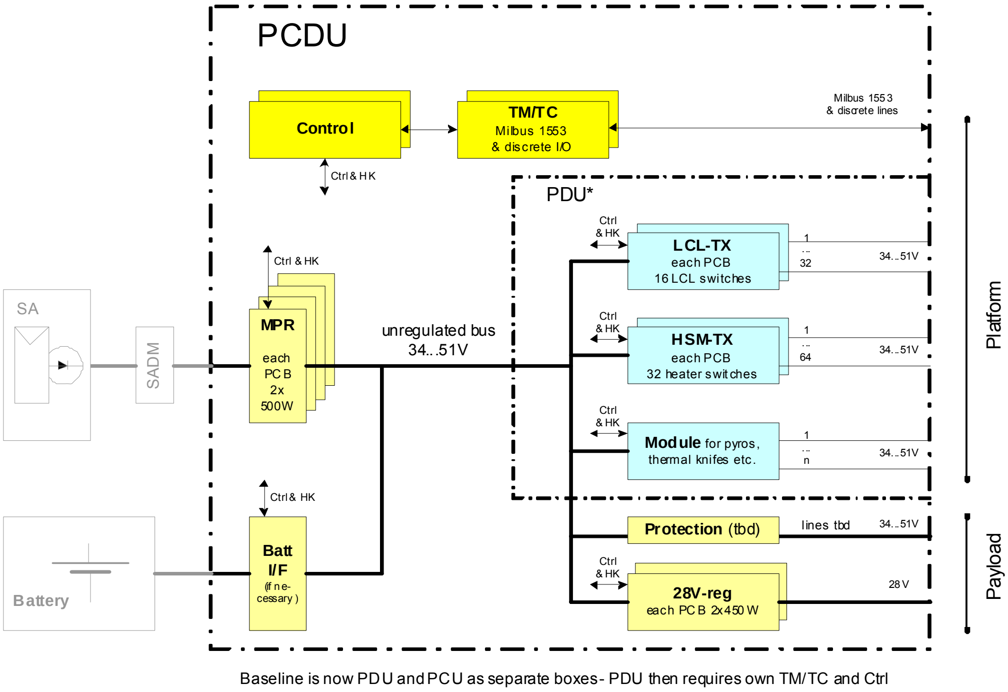


Figure 4.5-8 Block Diagram of LEO PCDU. Baseline is now PCU and PDU as separate boxes

EPS Summary

The following table summarizes the main properties of the LEO EPS:

Table 4.5-6 Summary of main evaluation and design results for the LEO EPS

Subject	Unit	Value	Remarks
General data			
sigma (fixed SA angle)	°	60	
Voltage main bus	V	33.6…50.4	for platform
Voltage sub bus	V	28 +/-1	for payload
Nom. power SA (BOL, 28°C)	W	3192.7	with MPR
Capacity battery	Wh	3110	
DoD battery	%	26	normal operation
Masses			
Mass SA	kg	37.8	2 full SGEO panels
Mass battery	Kg	31.9	
Mass PCU	Kg	18.6	MPR
Mass PDU	kg	12.1	MPR

OBDH Subsystem

The Onboard Data Handling Subsystem (OBDH) controls the platform functions and subsystems and provides the command and control interface to the payload support system (PSS). The platform electrical architecture together with the OBDH is shown in Figure 4.5-9 below.

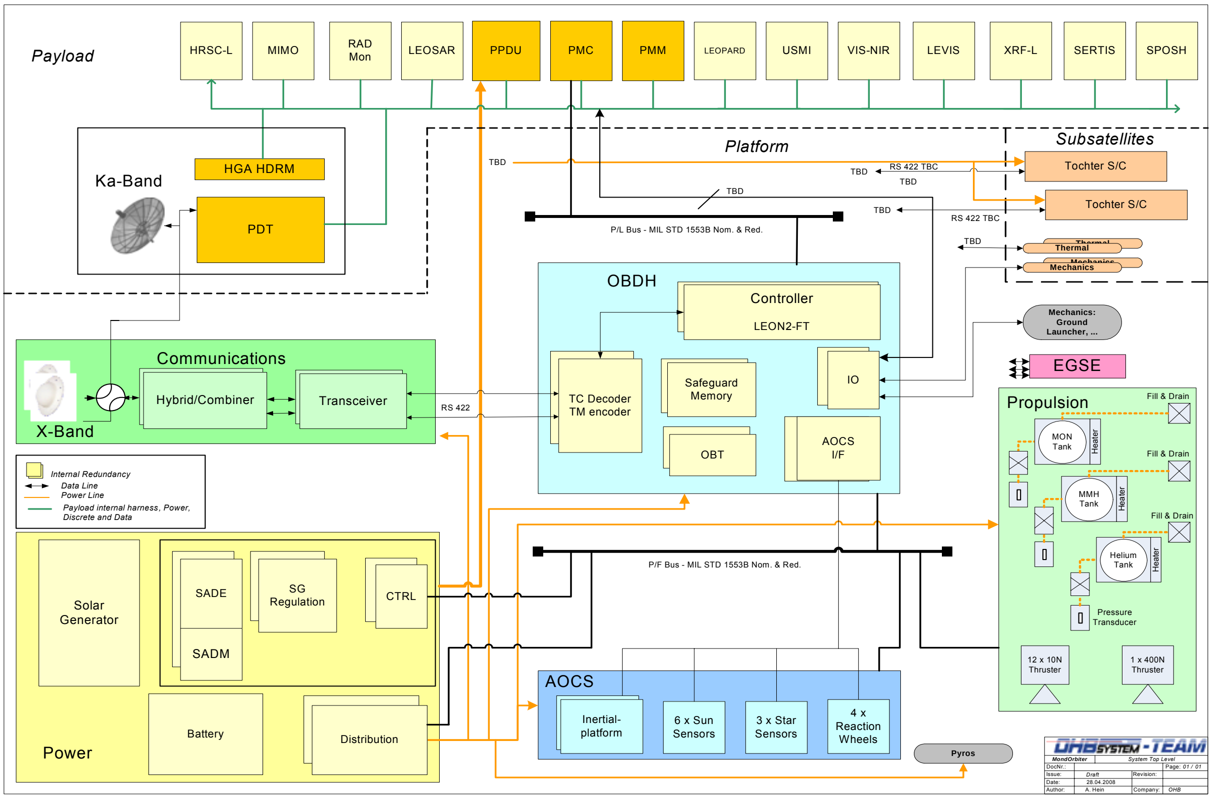


Figure 4.5-9 Principle of OBDH platform design

The core functionalities of the OBDH comprise:

- TC reception, decoding and TC handling
- TM transfer frame generation and encoding
- TC/TM provision to/from payload support section
- Processing capability for the on-board software
- Timing and synchronization management
- Control of the OBDH internal mass memory (TBC whether necessary)
- Platform mode control
- Payload support system mode control
- Power subsystem control
- Platform thermal control
- Attitude and orbit control
- Propulsion subsystem control
- Autonomy supervision and management (FDIR)
- Platform housekeeping TM acquisition and status monitoring
- Reception of payload support section HK TM packets from PMC and inclusion into platform TM stream

The OBDH function is physically implemented in the Data Management Unit (DMU), which is an internal redundant generic spacecraft management unit. The DMU has heritage from SGEO as well as from other missions such as Herschel/Planck, Aeolus, Galileosat or GAIA.

The DMU is built around a LEON II – FT microprocessor. Further hardware functionalities comprise the watch-dog together with the safeguard memory, the TM encoder and the TC decoder, the on-board time generation (OBT), I/O interface boards for the various subsystems and the DC/DC converters.

The DMU provides two redundant MIL busses, in accordance to MIL STD 1553B Notice2. One Milbus is used within the platform for the platform subsystems and equipments. The other Milbus is only interfacing to the PMC (Payload Management Computer) of the payload support section as centralized command and control interface. This centralized interface to the PSS ensures minimum interface complexity between platform and PSS and simplifies the interaction during development and AIV/AIT.

The OBT contains an OCXO (oven controlled crystal oscillator) which serves as time reference for the whole satellite. As for the Milbus, a centralized redundant pulse per second line is provided from the platform to the PMC which distributes the reference within the PSS.

The I/O interface boards provide discrete commanding and data acquisition capabilities and comprise UART channels and standard interfaces as e.g. high power commands, resistor sensors, analogue channels and bi-level digital channels.

The electrical interfaces to the subsatellites are limited to the release functions. The platform provides the actuation pulse for the release and receives the status TM from the separation bridge.

The advantages of the OBDH design can be summarized as follows:

- The DMU with heritage from many programs reduces development risk and cost.
- A minimized Platform/Payload interface reduces the effort for development and test.
- The MIL1553B Bus as remote control interface between platform and payload support section is a proven design and already used as standard interface for various space equipments and subsystems.

AOCS Subsystem

The most demanding requirements are (highlighted in bold):

- Pointing Accuracy: 0.1°
- Pointing Knowledge: 5“ (or 24.2407 µrad)
- Pointing Stability: 0.5“ / 0.56s (or 2.42470 µrad / 0.56s)

S/S Baseline Overview

The attitude and orbit control systems basically exists as closed control loop for the attitude and as corresponding switching of propulsion elements to support angular momentum management and orbit maintenance.

The associated guidance and control functions are carried out (together with other non AOCS related functions) by an AOCS software module on the board computer. Since for LEO the board computer is used for all system control functions, it is not specifically assigned as an AOCS element.

The specific AOCS elements are the sensors and actuators, these are:

a) 2 x Star Sensors (optional 3)

b) 4 x Gyros (later referred to as IMU)

c) Sun Presence Sensors (overall 12 solar cells, 2 on each of the 6 orbiter shells)

d) 4 x Reaction Wheels

e) 12 x 10 N Thrusters

f) 1 x 400 N Thruster

Propulsion Subsystem

A conceptual illustration of the propulsion module, featuring two fuel tanks, two oxidizer tanks and two pressurant tanks, is shown in Figure 4.5-10. The main engine nozzle protrudes from the underside of the module. The module is integrated into the satellite body and remains attached after burn-out.

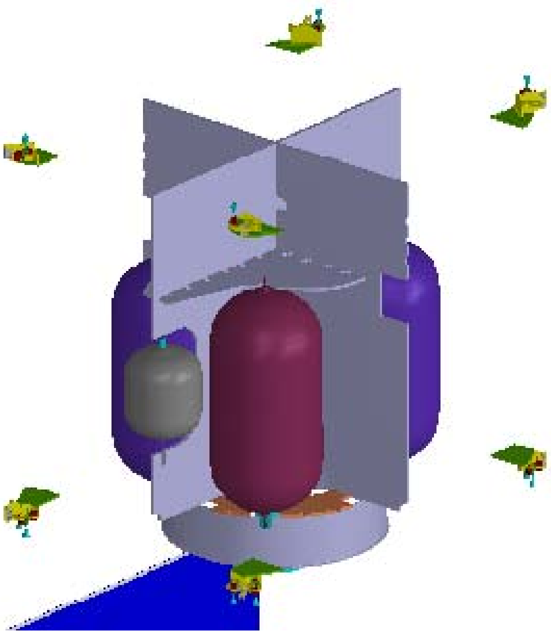


Figure 4.5-10 Liquid main engine module

A schematic block diagram is presented in Figure 4.5-11. The PPS is composed of redundant Helium, MON and MMH tanks, redundant 10N correction engines, and a single 400 N main engine. The usual complement of pyro valves, check valves and filters is provided, although not all are shown in the top-level diagram. There are also fill-and-drain valves, pressure transducers and trim orifices.

The functions of the propulsion subsystem are:

- storage and supply of pressurant
- separation of hypergolic fuel during the mission
- storage of fuel and oxidizer
- provision of fuel at controlled pressure for use during the main thruster manoeuvre
- provision of fuel in (mini) blow-down mode after the last use of the main thruster for use by the attitude control thrusters

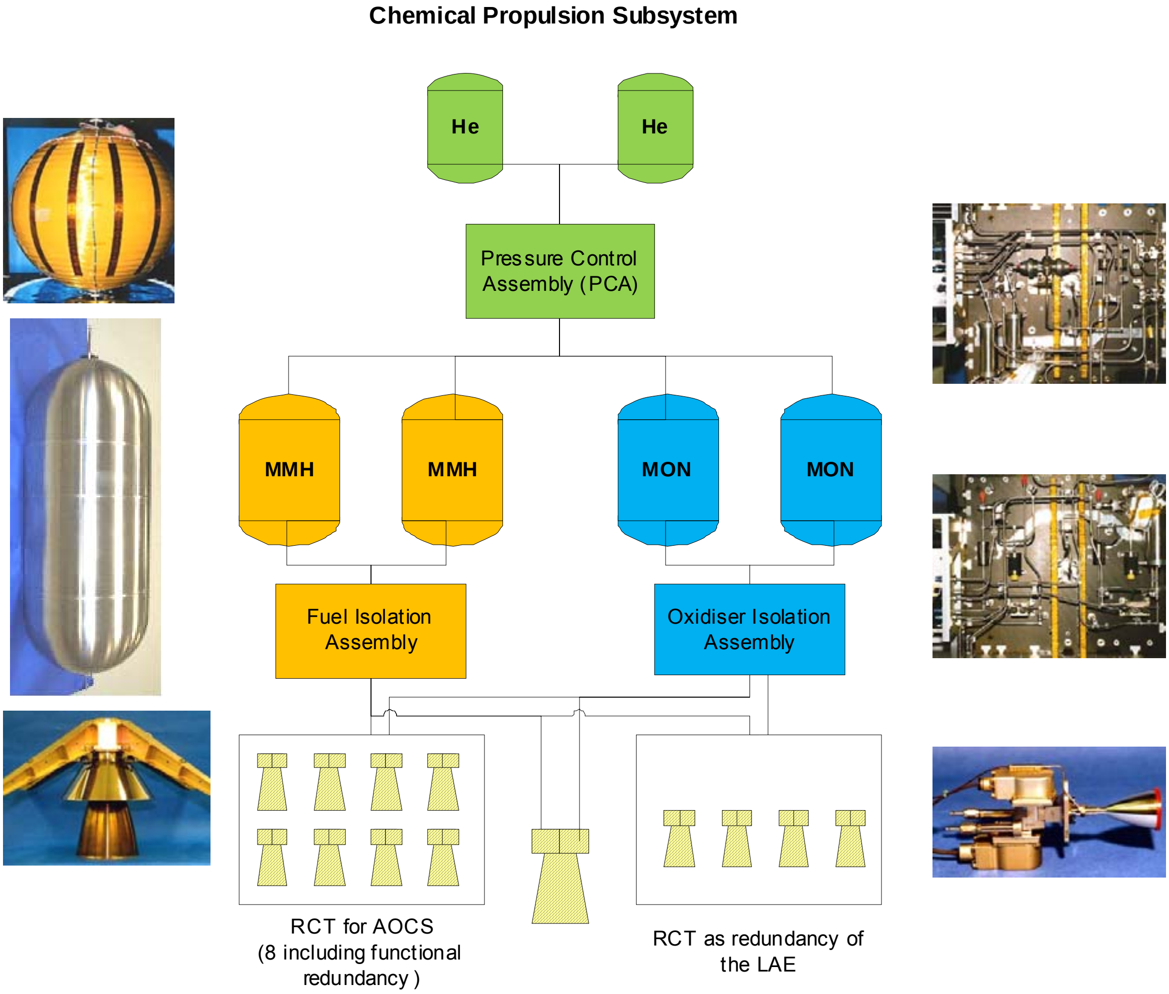


Figure 4.5-11 PPS Block diagram

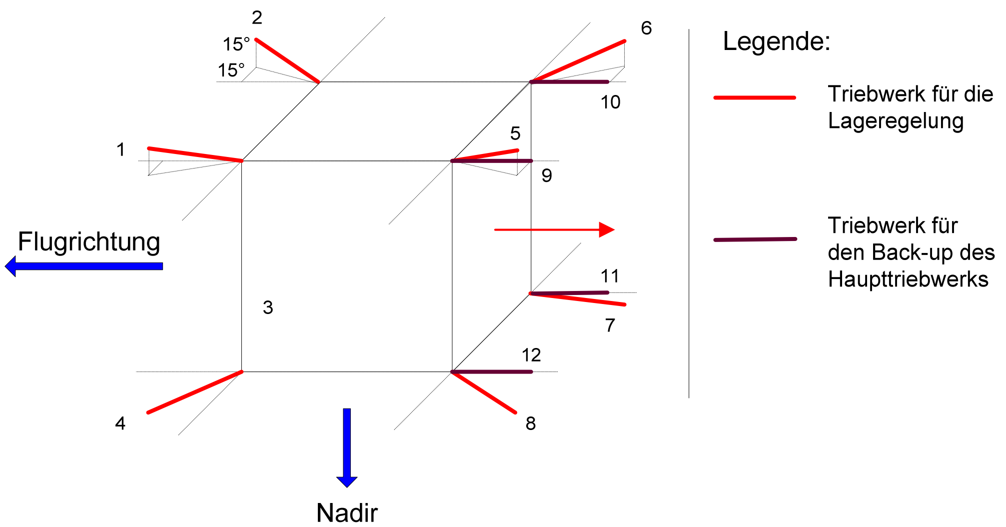


Figure 4.5-12 Thruster Configuration

Thermal Control Subsystem

S/S Requirement Analysis

The main function of the thermal control subsystem is to guarantee the required temperature ranges for all equipments in all mission phases. Whereas the main design drivers for the LEO TCS are as followed:

- high number of different instruments with different thermal requirements regarding thermal dissipations over time and temperature limits
- extreme changes of external thermal loads during the mission
- operational independence of instruments shall be possible, which leads to an individual and flexible TCS for each instrument.
- stringent temperature requirements related to the propulsion system and the battery

The Thermal Control System (TCS) is challenging through a large range of thermal loads from the changing thermal environment during the mission and the different thermal dissipations. To guarantee the required temperature ranges for all equipments in all mission phases a simple passive thermal control concept and design is proposed. The preliminary thermal control avoids the use of active elements like fluid loops or louvers. Only heaters with closed loop regulation based on majority voting of thermistors are proposed to be implemented in the TCS design.

In order to minimize the heater power consumption, different isolated cavities have been defined (battery, bus electronics units, tanks, instruments), with adequate regulation control and isolation.

The proposed thermal concept will be flexible enough to handle the interfaces with the instruments. The implementation of units will allow a robust design, using mostly existing components.

The LEO preliminary thermal design considers the following major features:

- The main energy balance of LEO will be governed by energy coming through the MLI, since the global dissipated power budget is low versus the associated external area. The orbiter will be sensitive to the MLI efficiency but the design will be flexible to adjust the global temperature level with modification of radiator areas after the thermal test.
- The spacecraft is entirely wrapped within high efficiency MLI to reduce the heat leaks to deep space and to avoid high heat fluxes to the orbiter from the sun and the moon. The only protruding elements will be radiators, thrusters, main engine, antennas, AOCS sensors and the sub-satellite interfaces.
- The solar panels back sides are painted white to reduce the albedo radiation from the moon.
- Bare parts of the launcher interface will be covered by a mix of Alodine and white paint coating to minimize fluctuation due to the solar flux.
- The radiators (total area: $\approx$ 4.0 m²) can be placed preferably on the shadow panel. Additional radiator area can be placed on the solar array side. The radiators will be covered with OSR.
- For heat distribution over the radiator areas and/or for coupling of units to the radiators heat pipes will be used.
- The internal housing surfaces will be black painted to couple radiatively all units, minimizing temperature gradients and taking advantage of the thermal inertia during transient phases.
- In order to minimize the heater power consumption and to achieve the temperature stability by some components like battery, tanks, fuel tubes etc. inner MLI isolation of the components in these cases will be used.
- Tanks are covered directly with MLI (VDA) and are individually thermally controlled.
- RCS lines, thrusters and valves will use classical conductive and radiative decoupling and heating techniques.
- HGA antenna will use a white painted coating on its front side and MLI on its rear side to

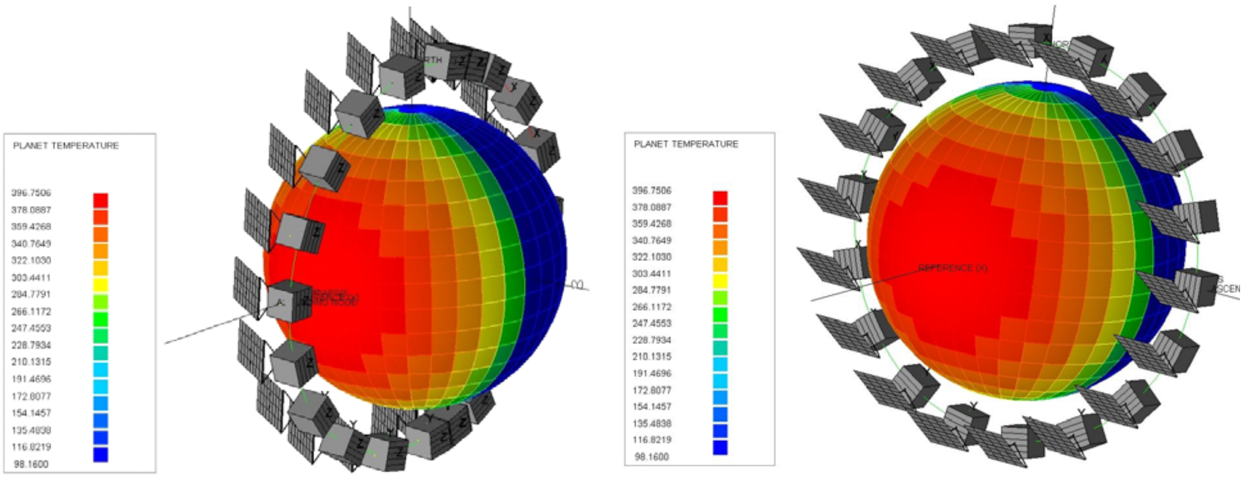
minimize temperature fluctuation and gradient.

- Star trackers will have a dedicated thermal control with heating lines and MLI tents, because the operational temperature requirement for good performance are always stringent.
- The thermistor controlled heaters are implemented with on board programmable software allowing a large flexibility during all mission phases.
- Appropriate redundancy is included for all heaters and thermistors to prevent single point failure in the thermal control function.

The main thermal design drivers for LEO are the different orbital moon environments. Two extreme orbital load cases have been analysed, these being the Noon-Midnight (0°) and the Dawn-Dusk (90°) orbit (see Figure 4.5-13).

The relevant environment data and assumptions are summarized as follows:

- Orbital Altitude = 50 km
- Orbit Inclination = 90°
- Radius of the Moon = 1737.4 km
- Distance to Sun = 149 597 870 km
- Infrared Emissivity of the Moon = 0.95
- Albedo of the Moon = 0.06 / 0.13
- Solar Absorptivity of the Moon = 0.94 / 0.87 (= 1.0 – Albedo)
- Solar Constant = 1420.0 W/m² (Moon perihelion)



Noon-Midnight Orbit Dawn-Dusk Orbit

Figure 4.5-13 Moon Orbits

<u>Telemetry, Telecommand and Ranging (TT&R) Subsystem</u>

S/S Requirement Analysis

According to the [AD2], following assumptions were made for the TT&R concept:

- A 16 m Antenna is the baseline for the whole mission. The RF-power and or data rate must be high enough to full fill all mission goals. For LEOP, normal operation and contingency it must be able to operate the spacecraft via the quasi omnidirectional coverage.
- The TT&R subsystem shall also be compatible to other ground stations (as a back-up), so international standards like CCSDS, ECSS must be followed.
- In addition to the spacecraft TT&R subsystem shall be compatible with common NASA standards.

S/S Baseline Overview

The layout consists of commercial off-the-shelf equipment which has heritage from former missions, so technical development risks can be minimized and only small changes must be done to fulfil the actual mission requirements. The layout is shown below.

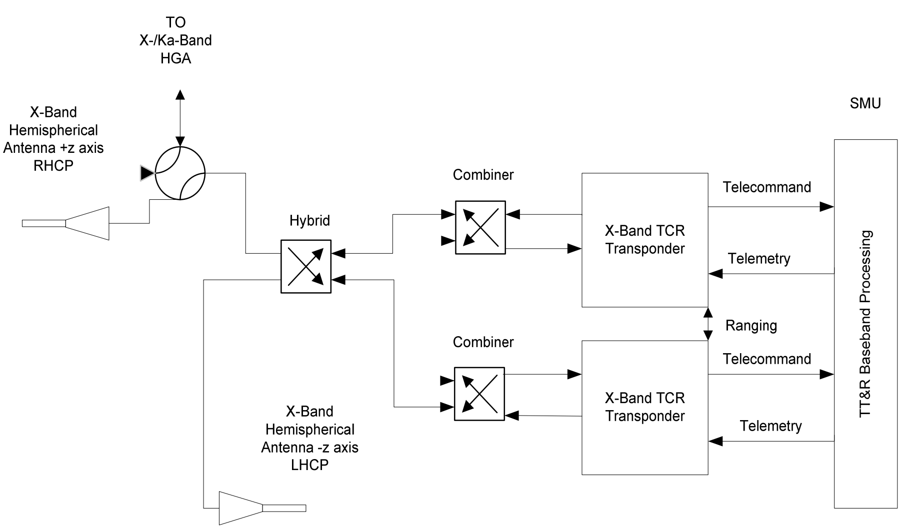


Figure 4.5-14 Layout LEO TT&R Subsystem

For the moment, only assumptions about the needed data rates are available. Once defined in further detail in the next project phase RF-Power and coding scheme will be reconsidered.

Structural Design

The LEO main satellite platform structural concept is based on the SGEO concept. In frame of the SGEO project one design goal was to provide a modular design providing high flexibility with low adaptation effort.

Some adaptations are necessary between the SGEO bus and the LEO platform, according to the different requirements mainly driven by different mass allocation.

The platform structure is composed of flat sandwich panels which facilitate the manufacturing and assembly of the structure subsystem. The platform structure panels are connected to each other with cleats or via direct panel-to-panel interfaces. The Core Platform Module, the Engine Module and the Payload Structure together form the bus. The separation of the bus structure in three sub-modules allows parallel integration and verification flow at system level – an important element for reducing the overall AIV time. The bus concept allows the integration and verification of the platform and payload equipment to a very high extent.

Figure 4.5-15 shows the exploded view of the platform structural elements. The antenna and instrument accommodation on the Nadir Panel, which is realized as not load carrying closure panel in SGEO, requires the further analysis. For the phase A conservatively a load carrying aluminium alloy was assumed. As mass reduction potential, an optimisation of fixation points with additional brackets connecting to the load carrying shear walls may be possible.

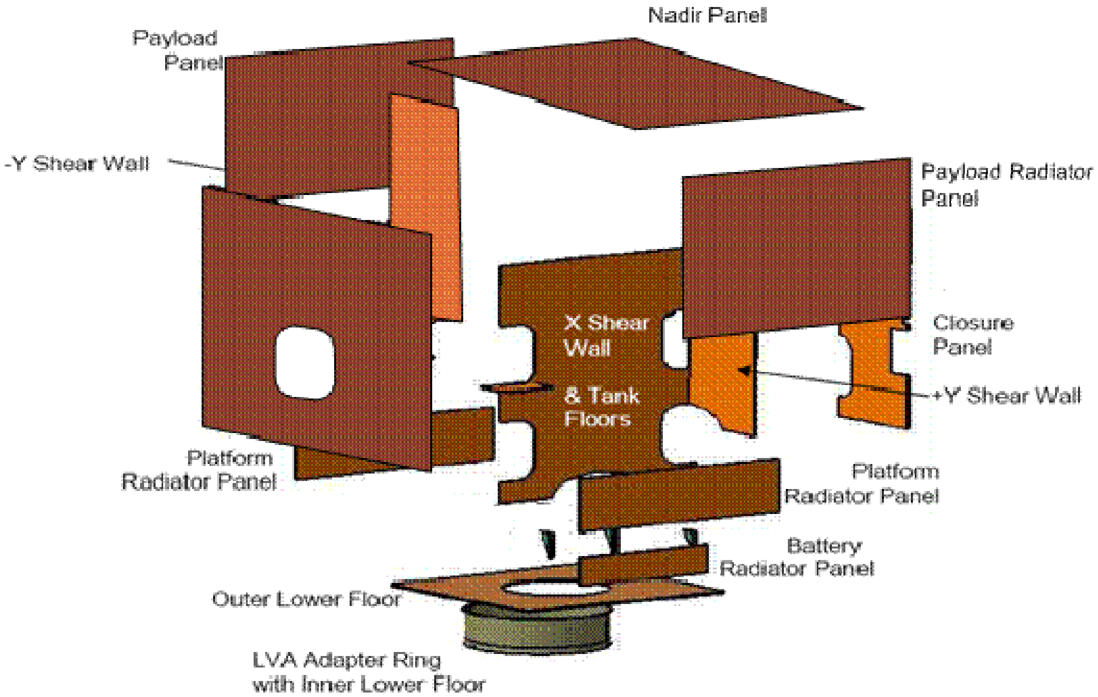


Figure 4.5-15 LEO bus structural elements.

4.5.2.3 Payload Support System

Rationale for a PSS

The payload support system (PSS) has been introduced to handle the interfaces between the 13 payloads (plus 2 sub-satellites) and the satellite platform. The work performed within the PSS shall result in clearly defined interface specifications for each of the items under concern.

One technical discipline concerned by the introduction of a payload support system is the electrical architecture. Since 12 payloads are foreseen to be accommodated on the main satellite, the control of these instruments becomes a significant task especially since many of them do not feature a dedicated instrument control unit. In order to reduce the software and interface complexity of the on-board computer of the platform, the payload management shall be handled by a dedicated payload management computer. All details involved in this topic are addressed in the next chapter.

The second technical issue is linked to the mechanical interfaces and compatibility. All of the payloads require adequate fields of view (for measurements and calibration) as well as suitable thermal conditions for heat radiation. All these requirements have to be considered when defining the accommodation of each item which has been assigned as a task for the PSS in close cooperation with the payload and the platform responsibles. The outcome of this activity will be presented in the chapter ‘instrument accommodation’.

Some additional statements concern the hardware assigned to the PSS and the modularity of the complete system. The only hardware which is physically provided by the PSS are some electronic boxes related to the payload management as shown in the next chapter. Moreover, some of the platform structure panels are defined as part of the PSS in a sense that the platform contractor has to provide them to the prime (as being responsible for the PSS). These are basically the panels where almost all of the payloads are mounted on. With these panels the pre-integration of the payloads can be done and a separate testing on sub-system level is possible. This modular approach is beneficial for the overall schedule since activities can be performed in parallel. It also helps to split responsibilities and to identify deficiencies in a sub-system early. It has to be mentioned that the platform contractor needs to manufacture additional dummy panels (with representative properties) for all panels assigned to the PSS such that a complete platform is available for testing.

PSS Functions and Design

Mechanical functions

The hardware of the PSS comprises the following items (see Figure 4.5-16):

- Panels +X, +Y, -Y and +Z,
- High gain antenna and payload data transmission electronics (orange boxes),
- PMC, PMM, PPDU (see *electrical functions* of this chapter),
- Sub-satellite release mechanisms (not yet designed and therefore not shown).

The structure panels as listed above are provided by the platform contractor. They are used to pre-integrate all payloads except for the sub-satellites (integrated on the platform) and 2 of the XRF instruments which are located on the –X panel. Functional testing shall be performed for all of these items on PSS level. This is possible when considering an extension harness for items which are not yet integrated and therefore placed somewhere beside the panels during testing.

The complete payload data transmission chain including the high gain antenna (with deployment and steering mechanisms) is part of the PSS functions as it can be seen from above hardware list.

The release of the sub-satellites is also a function provided by the PSS. The design of the release mechanism has to consider the separation dynamics between the sub-satellites and the spacecraft in order to avoid collision and it is therefore a clear system task.

Information about the payload data management is provided hereafter.

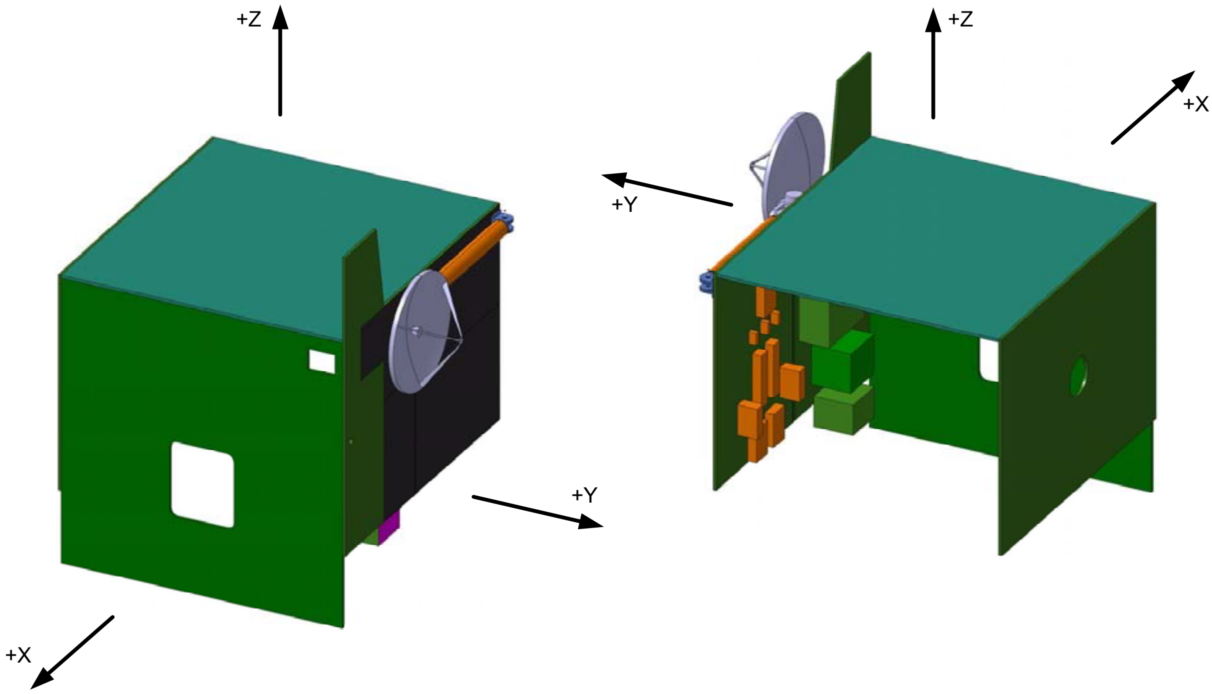


Figure 4.5-16: LEO PSS constituents

Electrical functions

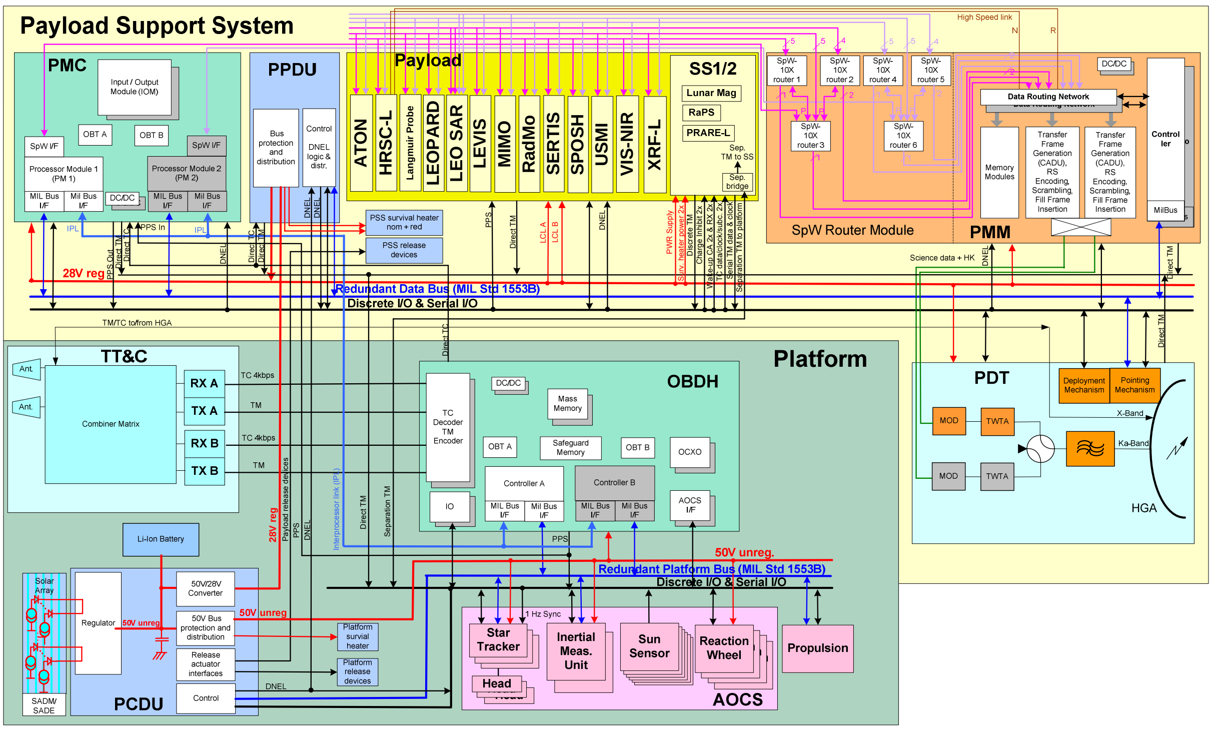


Figure 4.5-17: LEO Electrical Architecture

The LEO electrical architecture consists of two major modules driven by the functional split, the platform and the payload support system. The platform is described in detail in another chapter. The platform provides the power generation, its own on-board computer with AOCS control and the X-band TT&C (Telemetry & Telecommand) communication via LGA. The payload support system interfaces to all instruments and provides a payload management computer (PMC) for instrument control, the payload power distribution unit (PPDU), the payload mass memory (PMM) and the payload data transmission (PDT) for downlinking of the science data. The electrical PSS functions and the electrical design of the PSS are shown above.

The power bus voltage in the PSS has been set to 28V regulated, since 28V regulated or unregulated was preferred by most of the instrument providers. Since 50V is the standard voltage for the SGEO platform, a voltage converter can provide a regulated bus voltage which improves the efficiency of the connected DC/DC converters of the PSS equipments. The PPDU provides nominal power plus survival heater power for the payload equipments/instruments. Both types of power lines are protected by LCLs. In addition, the DNEL (disable non-essential loads) signal received by the platform is also distributed to the payload equipments/instruments.

The communication within the payload support section is based on SpaceWire, Milbus and discrete TM/TC. SpaceWire has been selected for the instruments communication, since Milbus was not favoured by the instrument providers due to its power consumption and operational complexity. The Spacewire interfaces are used for telecommand, telemetry and mostly also for the science data (see below). Therefore, a cascaded router functionality is implemented to provide failure redundancy and routing of the packets to the corresponding addressees.

The communication between PMC, PPDU, PMM and HGA pointing mechanism is based on a redundant Milbus (MilStd 1553B). Alternatively, RS422 (UART) interfaces could be used for communication. This will be investigated further since it would allow complete deletion of the Milbus within the payload section providing the advantage of less communication protocols to be defined.

The subsatellites are also interfacing to the payload support section except for the separation bridge TM which is directly connected to the platform to account for any potential impact of the separation towards the AOCS.

A time reference distribution is foreseen within the payload section in two manners, either via the SpaceWire time synchronization procedure as defined in ECSS-E-50-12A or alternatively via a discrete hardware pulse per second (PPS). The instrument selection of one or the other method depends on the requirements on the timing accuracy. The OBT (on-board timer) of the PMC is synchronized to the OBT in the platform on-board computer via a dedicated PPS line. The PMC distributes the PPS to the units and instruments of the payload section. The OBT of the platform on-board computer is synchronized to an OCXO allocated within the platform.

Payload Communication Concept

As said before, the payload communication concept is based on a SpaceWire architecture for TM, TC and science data. For this purpose, a router functionality has to be implemented in order to provide the packets to the correct destinations between instrument, PMC and mass memory. Due to the high rates, the science data of HRSC-L are routed via a dedicated redundant high speed link (e.g. Aeroflex UT 54 LVDS 217/218) to the mass memory. Redundancy of the SpaceWire connections is foreseen, i.e. each instrument will have two SpaceWire interfaces irrespective of the instrument internal redundancy concept. Thus it is guaranteed, that a router failure will not lead to loss of a complete series of instruments. Cross-coupling of the redundant router chains is performed within the data routing network of the mass memory.

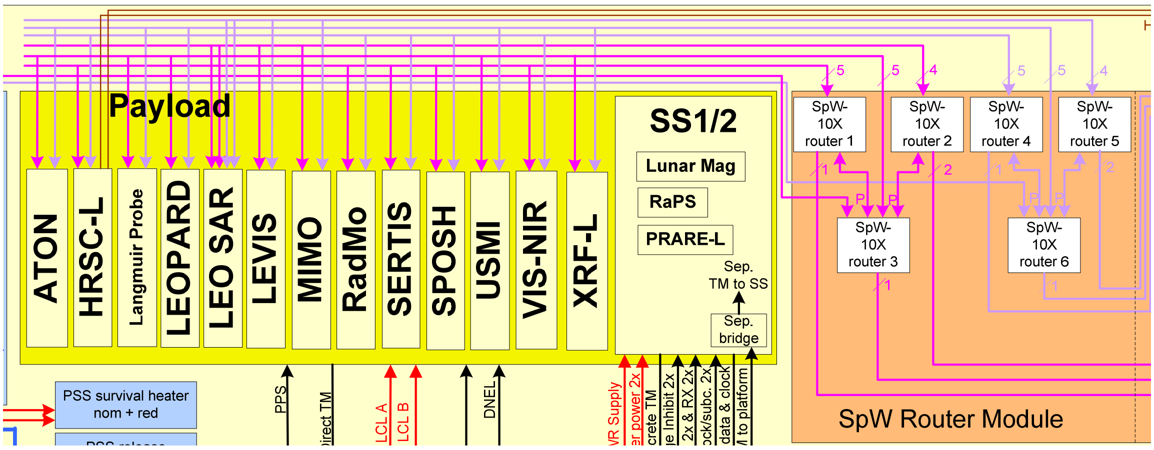


Figure 4.5-18: LEO Payload Communication Concept

The instrument allocation to the different routers is based on the peak data rates and the different observation modes, together with the limit of the router physical data rate per port of 200Mbps or net data rate of around 150Mbps. For the optical instruments, additionally the varying data rate due to the altitude over ground is considered (here worst case is assumed to be 25km), however the resulting impact on the data rate towards the router is limited (VISNIR as the driving instrument is limited to one SpW line). All instruments can be operated in parallel although this scenario is not considered realistic.

Table 4.5-7 Router ports allocation and router load

SpW Capacity Assessment (nominal or red.)								
SpW line capacity / Mbps	150	Router 1	Router 2	Router 3	Router 1	Router 2	Router 3	Remarks
	SpW lines	SpW lines	SpW lines	SpW lines	Data rate	Data rate	Data rate	
PMC	1			1	0	0	0	
ATON	1			1	0	0	64	Should be no driver
HRSC-L	1	1			0	0	0	
Langmuir Probe	1			1	0	0	0	
LEOPARD	1			1	0	0	0,2	
LEOSAR	2		2		0	203	0	
LEVIS	1		1		0	25	0	Should be no driver
LunarMag: Daughter	0				0	0	0	
MIMO	1		1		0	0,016	0	
PRARE-L: Daughter	0				0	0	0	
RadMo	1	1			0,116	0	0	
RaPS: Daughter	0				0	0	0	
SERTIS	1	1			0,8	0	0	
SLR	0				0	0	0	
SPOSH	1	1			0,0093	0	0	
USMI	1			1	0	0	25,8	
VIS-NIR	1	1			150	0	0	VISNIR limited to one SpW line, i.e. 150Mbps
XRF-L	1			1	0	0	0,01016	
Sum input lines	15	5	4	6	151	228	90	
Output lines if per observation mode	2							Only one observation mode at a time
Total lines	17							
Amount of routers	3							
Output lines if all instruments active	4	1	2	1				All instruments may be on at the same time
Total lines	19	6	6	7				
Amount of routers	3							

Currently, the SpaceWire router module is located within the mass memory. This needs to be further iterated, since in this case communication with the instruments can only be done when the mass memory is switched on. A decentralized SpW router module could be possibly advantageous in order to reduce the cable lengths between router module, instruments and mass memory.

Concerning router availability, the SpaceWire router has now been successfully qualified and is available as ASIC from ATMEL. It will also be used on the Bepi Colombo mission for communication with the instruments.

Payload Management Computer

The PMC controls the complete payload support system including TM acquisition and supervision, scheduling, time synchronization and FDIR within PSS.

The payload management computer consists of redundant processor modules with Milbus and SpaceWire interfaces together with redundant on-board timers. In addition, input/output modules will serve the discrete interfaces as low or high power commands, RS422 communication links, thermistor and analogue channels acquisition and digital status acquisition.

Reconfiguration modules are not foreseen within the PMC, since in case of PMC anomalies the platform OBDH will be the next hierarchical instance within the FDIR to initiate any recovery action.

The SpaceWire lines interface with the router module for communication with the instruments.

The interprocessor link between PMC and platform on-board computer is based on a redundant Milbus. The PMC will receive payload section mode commands from the platform on-board computer together with Housekeeping TM which is included in the Ka-band payload data stream to ground. The PMC sends its housekeeping TM to the platform for inclusion into the X-band TM stream. In addition, ancillary data like the S/C state vector (orbit position and attitude) are provided by the platform on-board computer to the PMC. The PMC distributes this information to the instruments accordingly.

Payload Power Distribution Unit

The LEO PPDU (Payload Power Distribution Unit) receives 28V regulated from the platform and distributes it to the payload support system users via LCL protected and switched lines. For the instruments, cold redundant sets of power outlets are provided. In case there is no instrument internal redundancy, the redundant power lines are Or-ed with diodes in the instruments. Redundant power lines for the survival heaters are also provided by the PPDU. Several survival heater lines (typically 4) are protected by one LCL. This means that the PPDU needs to be active also during survival mode. Due to their simple implementation and without the need for the PMC to be active for thermal control, thermostat controlled survival heater are preferred.

In case of battery undervoltage, a DNEL (Disable Non-Essential Loads) signal is received from the platform and distributed via the PPDU to the affected hardware plus LCL switch-off of the affected power lines 20sec (TBC) after an active DNEL signal.

Thus, the PPDU ensures that the power interface to the platform is kept to a minimum and that a rather self-standing payload support section is defined.

The PPDU supports the satellite distributed single point grounding (DSPG) where the primary power returns will be grounded to the spacecraft structure at the star point within the platform (the negative side of the main bus capacitor).

Payload Mass Memory

The PMM (Payload Mass Memory) shall collect the science data from the instruments together with HK TM from instruments, PSS and platform and store this data during non-visibility periods of the ground station (nominally one ground station in Weilheim). When the ground station is visible, the data are read from the memory, formatted and encoded, scrambled and sent to the Ka-band payload data transmission.

The data routing network provides cross-strapping between the different input and output modules and interfaces to the mass memory modules and the formatting module. The PMM controller commands the correct switching of the data paths.

Compression is foreseen to be performed within the instruments.

Due to the large memory size (see below), NAND flash technology is selected which provides much higher storage density compared to SDRAM. This technology has so far not been flown, but radiation tests have been performed successfully and the required mitigation methods are known. NAND flash memories are limited due to their technology in the amount of write cycles. Typically, around 100000 write cycles are possible, in combination with dynamic block addressing even more. The LEO write cycles are far below this limitation with the assumptions that the number of ground contacts is equal to the number of write cycles and that max. 2000 ground contacts occur per year.

The sizing of the mass memory is done according to the ground station visibility, based on an average instrument input rate of 25Mbps (including CCSDS source packet formatting) and one ground station located at Weilheim. The contact profile uses 10° minimum elevation and 20sec contact build-up margin. The memory output data rate sent to the input of the formatting module is set to 200Mbps. The data rates have not been changed compared to the phase 0 study since design limits are reached for TWTA output power in Ka-band and the HGA. It was also considered reasonable to avoid any additional growth of the mass memory.

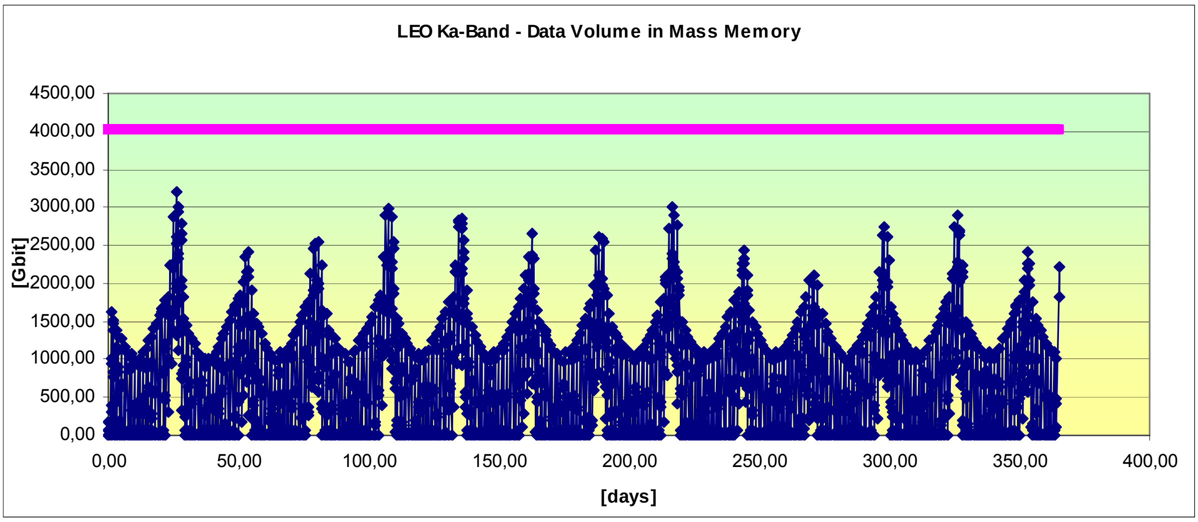


Figure 4.5-19 Data Volume in Mass Memory

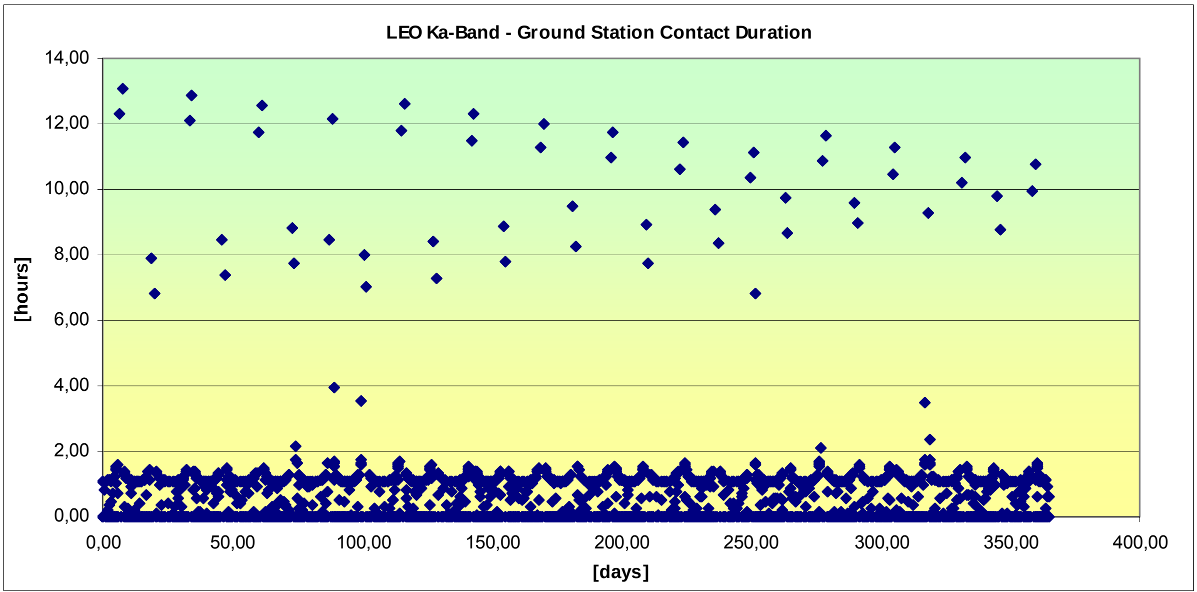


Figure 4.5-20 Ground station contact durations

This gives a minimum required size without margin of 3.2Tbit and 4Tbit with 25% margin. Per mass memory board a 1Tbit storage capacity is assumed. For redundancy, one additional independent board is foreseen, such that 5 boards are required. This gives in total a BoL mass memory size of 5Tbit.

The Ka-band availability has been assumed to be 90% for reasonable link sizing (see below in the PDT section). As a result of this, a handshake mechanism has to be implemented via the TM/TC link which indicates the memory areas which can be released after successful transfer. The availability also affects the sizing of the mass memory since 10% margin has to be allocated, i.e. instead of 3.2Tbit then 3.5Tbit are required, which is covered by above mass memory size.

The encoding foreseen in the formatter of the mass memory is Reed Solomon (RS) 255/223 and ¾ convolutional. It is selected for high power efficiency at low error rate together with moderate bandwidth expansion.

Payload Data Transmission

The payload data transmission operates in Ka-band (25.5 to 27GHz) due to its high data rate and corresponding large occupied bandwidth. The PDT modulates the instrument data stream on the RF carrier by QPSK modulation, amplifies with a TWTA, filters the output signal and transfers it via a 0.75m diameter high gain antenna (HGA) to the ground. Circular polarization is foreseen. The transmitting chains are cold redundant and a WG switch in the RF path selects the active chain before the common output filter. The HGA is a two axes steerable antenna to provide continuous pointing to a ground station once visible. The pointing mechanism provides omni-spherical coverage. In order to avoid shading of the HGA by the rotating solar array, the HGA is mounted on a deployable boom.

In addition to the Ka-band downlink, an X-band up- and downlink is foreseen for the HGA. This gives for nominal operations higher rates for the X-band TC and especially also for X-band TM links.

The link budget is based on a ground station Ka-band antenna with a diameter of 16m, minimum elevation of 10°, 90% availability giving 3.3dB atmospheric attenuation, a bit error rate of 10^{-9}, encoding of RS 255/223 and ¾ convolutional. The moon noise temperature has been assumed with 249K at full moon based on GSOC measurements in S-band at Neustrelitz. This gives a system noise temperature of 696K as calculated in the table below. Punctured convolutional encoding with moderate bandwidth increase is implemented to compensate for the high system noise temperature. The resulting bandwidth is approx. 218MHz (99%). The Ka-band TWTA has an output power of 60W RF.

Higher availability figures would significantly increase the atmospheric loss (e.g. 98% gives already 6.3dB total atmospheric loss) and then push the Ka-band feasibility beyond its design limits for such high data rates at this distance.

Instrument accommodation

There is a rather long list of payloads (13 instruments plus 2 sub-satellites) to be accommodated and it is therefore helpful to identify some major constraints at first. The flight configuration without payloads is shown in Figure 4.5-21. The panel on which the flight direction is indicated corresponds to the top panel during the launch phase. The first decision which was taken is that on this panel the two sub-satellites will be accommodated. This choice is beneficial for the transfer of loads to the satellite structure as well as for the separation scheme since the sub-satellites are already well oriented with respect to their later flight direction.

Prior to explaining the other accommodation decisions, a few statements about the thermal environment are made. Over one year, the spacecraft is exposed to varying sun incidence angles such that, in general, for each panel changing sun exposure conditions have to be considered. However, the mission profile has been established such that due to a yaw flip manoeuvre every half a year the so-called shadow panel is never exposed to direct sun light. This panel is situated opposite to the solar array fixation panel where, accordingly, sun illumination is always provided. At a first glance, the shadow panel is the natural first choice for installing radiators of the thermal control system but it must not be forgotten that the infrared heat flux from the moon is a significant heat source. Hence, when designing the thermal control system the Zenith panel which is sun exposed but not loaded with external infrared heat flux could also be considered for radiator areas.

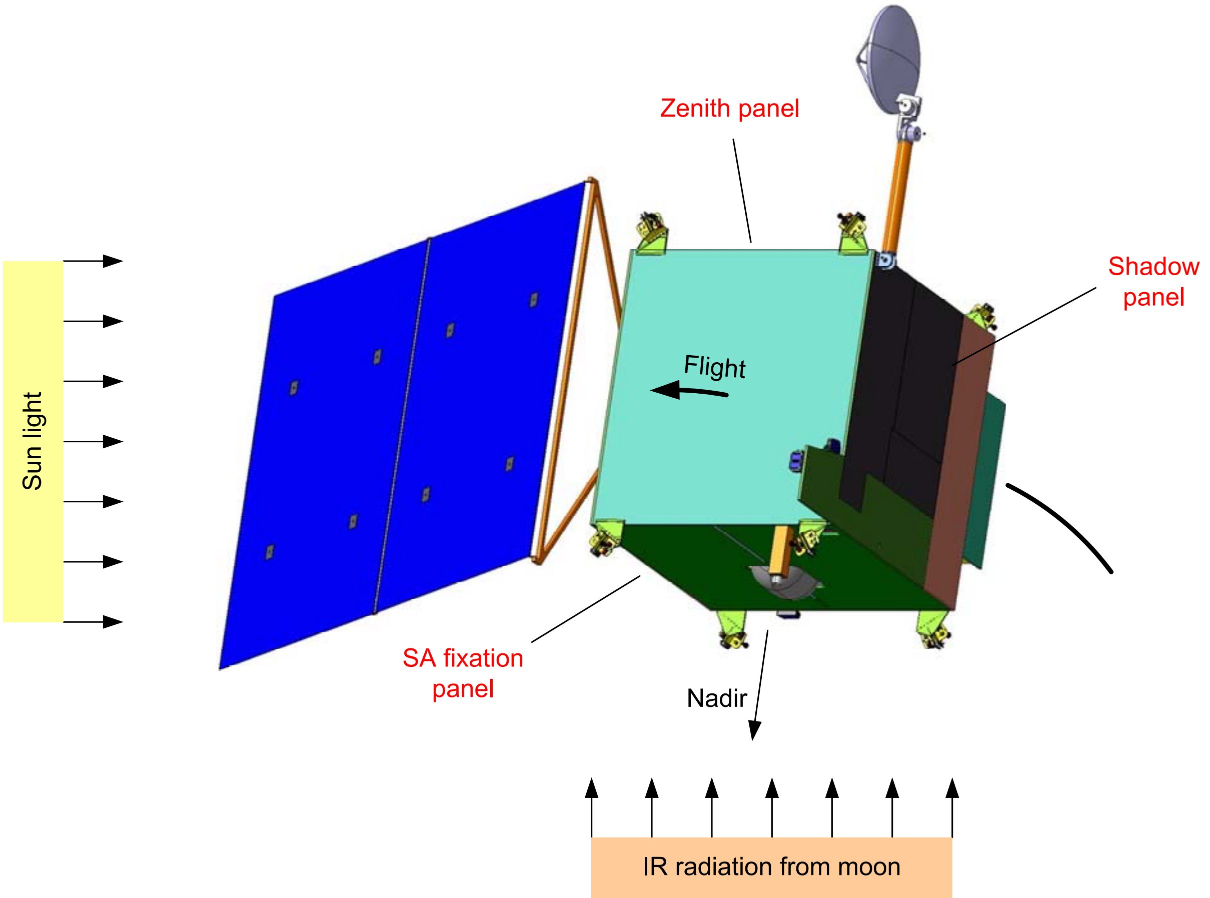


Figure 4.5-21 Thermal environment in flight configuration

With the explanations of the thermal environment it can be started to accommodate the payloads. For the Nadir looking instruments with a significant demand for heat rejection it is clear that the edge between the Nadir panel and the shadow panel is a preferred location. There, the observation conditions as well as the possibility for heat dissipation are optimum. The other edges of the Nadir panel (towards flight / anti-flight directions and towards S/A fixation panel) offer equally good observation conditions but the capabilities for heat rejection are worse. Hence, the following instruments are placed in the preferred location:

- HRSC-L,
- SERTIS,
- USMI,
- VIS-NIR.

The next items are the LEOSAR and the MIMO instruments which both need a certain field of view towards Nadir. The MIMO is composed of 5 different instruments each covering a dedicated frequency, i.e. 7 GHz, 24 GHz, 60 GHz, 180 GHz and 557 GHz. The most challenging one is the 7 GHz antenna since this corresponds to the highest wave length out of the selected frequencies. Hence, this results in the biggest antenna size which is about 1 m^2. Similar dimensions apply to the LEOSAR with about 1.5 m x 0.8 m. For both antennas the Nadir panel as well as the adjacent side panels have been traded as possible accommodation locations. With the given dimensions it is the most convenient solution to fix these antennas on the Nadir panel. In their operational configuration they fit into the available Fairing volume of Soyuz ST without the need of a deployment mechanism.

The rest of the payloads are rather compact, in general. Nevertheless, for each item the fulfilment of the field of view requirements has been checked. They can all be accommodated but it must be stated that the available volume on the spacecraft to comply with all requirements is rather stringent.

The accommodation of all payloads on the panels assigned to the payload support system is shown in Figure 4.5-22 (external accommodation) and Figure 4.5-23 (internal accommodation of equipment).

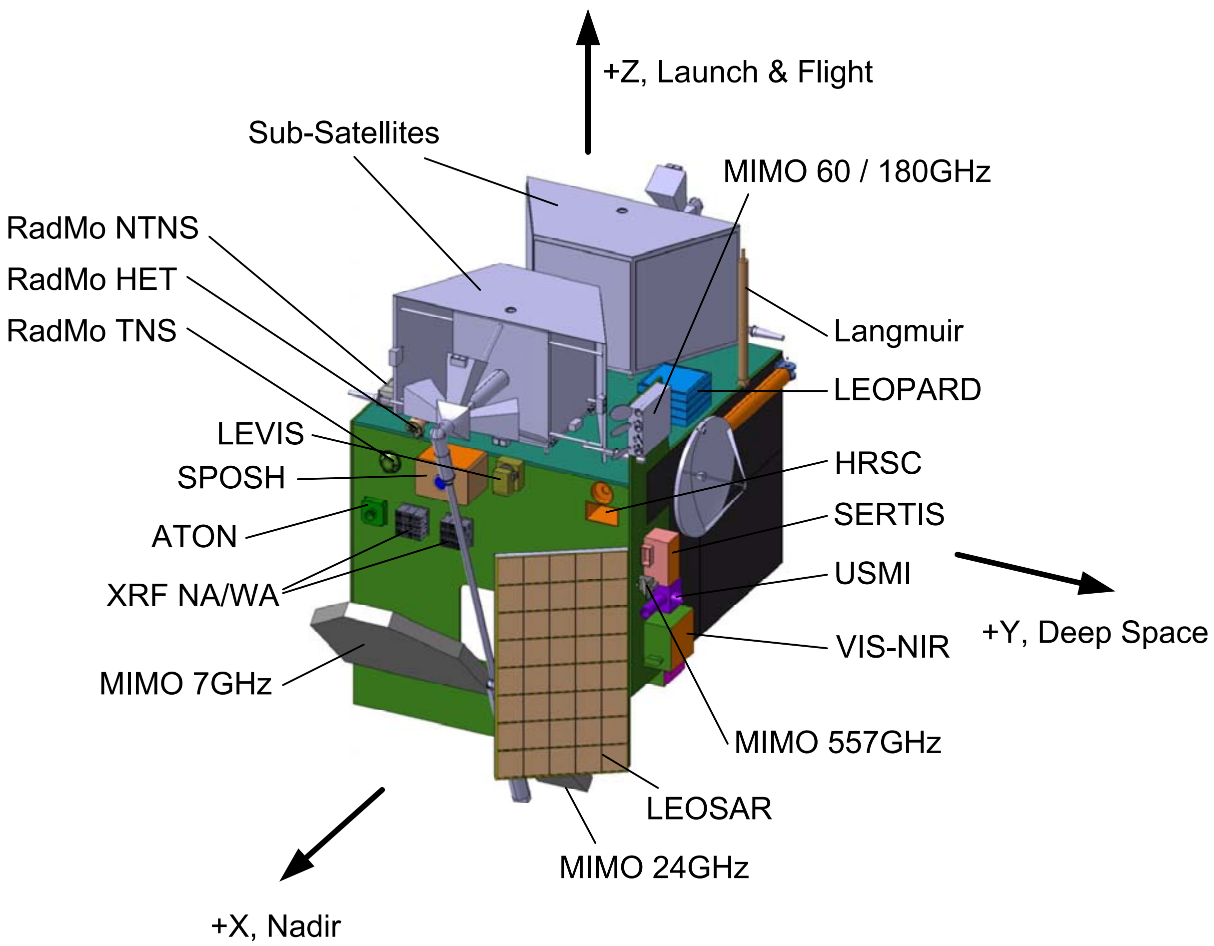


Figure 4.5-22 Isometric view of payload support system, external payloads

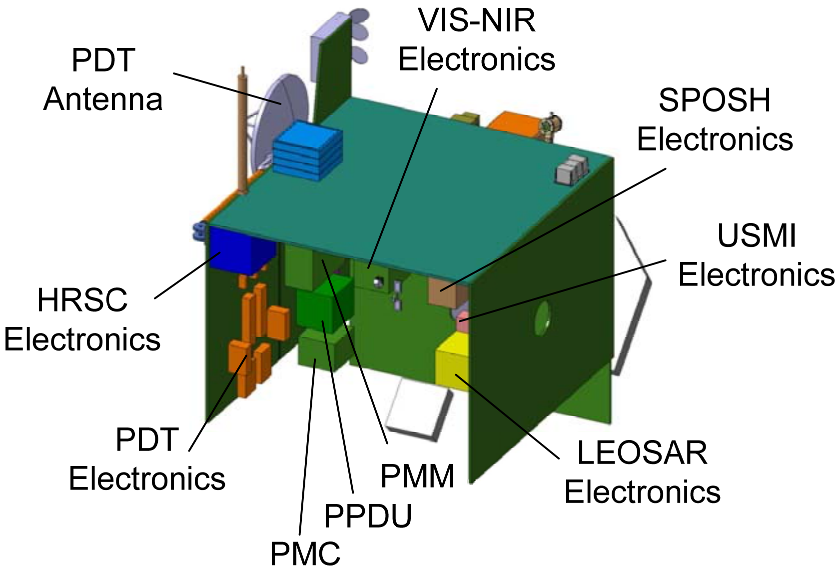


Figure 4.5-23: Isometric view of payload support system, internal equipment

PSS Budgets

Table 4.5-8: PSS Mass Budget

Subsystem	Item	No. of units	Basic mass [kg]	Mass w/o margin [kg]	Margin [%]	Mass incl. margin [kg]	Remarks, References
Payload	HSRC-L	1	25,0	25,0	20	30,0	
	Langmuir probe	1	0,9	0,9	10	1,0	
	LEOPARD	1	6,5	6,5	20	7,8	
	LEOSAR	1	20,5	20,5	20	24,6	
	MIMO	1	26,4	26,4	20	31,7	
	RadMo	1	7,4	7,4	20	8,9	
	SERTIS	1	3,5	3,5	20	4,2	
	SPOSH	1	2,5	2,5	20	3,0	
	USMI	1	7,0	7,0	20	8,4	
	VIS-NIR	1	10,0	10,0	20	12,0	
	XRF-L	1	12,6	12,6	20	15,1	
	LEVIS	1	2,1	2,1	20	2,5	
	ATON	1	1,0	1,0	20	1,2	
	Total Payload			**125,3**	**20**	**150,3**	
PSS	PPDU	1	11,2	11,2	10	12,3	Only LCL + heater
	PMC	1	8,4	8,4	5	8,8	No RM
	PMM	1	16,5	16,5	10	18,2	5Tbit BoL flash
	HGA assembly	1	38,4	38,4	20	46,1	
	Ka-Transmitter	2	3,4	6,9	20	8,2	
	Switches, filters & cables	set	4,5	4,5	10	5,0	
	Total PSS			**85,9**		**98,6**	
Total Payload System				**211,25**	**17,82**	**248,89**	
Subsatellites	Subsatellites	2	120,9	241,8	3	250,0	
	Propellant	2	7,0	14,0	0	14,0	Margin included in basic mass
	Total Subsatellites			**255,8**	**3**	**264,0**	

N.B: PS harness is not included in the above numbers and is to be included in Platform harness mass.

Table 4.5-9 PSS Power Budget (PDT off)

LEO Power Budget

PSS — Observation Mode:

Without Payload Data Transmission

Element	Unit	No. of active Units	Unit Margin (%)	Ad Unit Power (W)	A'd Unit Power (W)	A'n Unit Power (W)	Bn Unit Power (W)	B'n Unit Power (W)	Cd Unit Power (W)	C'd Unit Power (W)	C'n Unit Power (W)	Dd Unit Power (W)	Dn Unit Power (W)
Payload				**411,7**	**261,7**	**247,7**	**312,3**	**240,2**	**288,4**	**254,2**	**240,2**	**254,2**	**240,2**
	ATON	1	20%	3	3	3	3	3	3	3	3	3	3
	HRSC-L	1	20%	225	75	75	68	68	68	68	68	68	68
	Langmuir Probe	1	10%	5	5	5	5	5	5	5	5	5	5
	LEOPARD	1	10%	16	16	16	16	16	16	16	16	16	16
	LEOSAR	1	20%	31	31	31	103	31	31	31	31	31	31
on during transfer	LEVIS	1	10%	4	4	4	4	4	4	4	4	4	4
	LunarMag: Daughter	0	5%	0	0	0	0	0	0	0	0	0	0
	MIMO	1	30%	54	54	54	54	54	54	54	54	54	54
	PRARE-L: Daughter	0	20%	0	0	0	0	0	0	0	0	0	0
on during transfer	RadMo	1	20%	11	11	11	11	11	11	11	11	11	11
	RaPS: Daughter	0	20%	0	0	0	0	0	0	0	0	0	0
	SERTIS	1	10%	15	15	15	15	15	15	15	15	15	15
	SLR	0	30%	0	0	0	0	0	0	0	0	0	0
	SPOSH	1	20%	3	3	10	10	10	3	3	10	3	10
	USMI	1	30%	10	10	10	10	10	34	10	10	10	10
	VIS-NIR	1	20%	5	5	5	5	5	15	5	5	5	5
	XRF-L	1	30%	30	30	9	9	9	30	30	9	30	9
Thermal				**100,0**	**100,0**	**100,0**	**100,0**	**100,0**	**100,0**	**100,0**	**100,0**	**100,0**	**100,0**
PDHT				**55,0**	**55,0**	**55,0**	**55,0**	**55,0**	**55,0**	**55,0**	**55,0**	**55,0**	**55,0**
	PMM	1	10%	40,0	40,0	40,0	40,0	40,0	40,0	40,0	40,0	40,0	40,0
	Modulator	1	20%	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0
	TWTA	1	20%	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0
	HGA Pointing Mechanism	1	20%	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0
PMC				**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**
	PMC	1	5%	20,0	20,0	20,0	20,0	20,0	20,0	20,0	20,0	20,0	20,0
Power				**47,3**	**39,0**	**38,3**	**41,8**	**37,8**	**40,5**	**38,6**	**37,8**	**38,6**	**37,8**
	PPDU	1	10%	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0
	Conversion loss 50V-->28V	0,0%	5%	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0
	Harness Loss	3,5%	5%	20,5	15,3	14,8	17,1	14,5	16,2	15,0	14,5	15,0	14,5
	Distribution Loss	2,0%	5%	11,7	8,7	8,5	9,7	8,3	9,3	8,6	8,3	8,6	8,3
Total				**634**	**476**	**461**	**529**	**453**	**504**	**468**	**453**	**468**	**453**
Margin													
Units margin				104	73	68	82	67	81	72	67	72	67
System margin			10%	74	55	53	61	52	59	54	52	54	52
PSS total with margin	W			**812**	**604**	**582**	**672**	**572**	**644**	**594**	**572**	**594**	**572**

Table 4.5-10 PSS Power Budget (PDT on)

LEO Power Budget

PSS — Observation Mode:

With Payload Data Transmission

Element	Unit	No. of active Units	Unit Margin (%)	Ad Unit Power (W)	A'd Unit Power (W)	A'n Unit Power (W)	Bn Unit Power (W)	B'n Unit Power (W)	Cd Unit Power (W)	C'd Unit Power (W)	C'n Unit Power (W)	Dd Unit Power (W)	Dn Unit Power (W)
Payload				**411,7**	**261,7**	**247,7**	**312,3**	**240,2**	**288,4**	**254,2**	**240,2**	**254,2**	**240,2**
	ATON	1	20%	3	3	3	3	3	3	3	3	3	3
	HRSC-L	1	20%	225	75	75	68	68	68	68	68	68	68
	Langmuir Probe	1	10%	5	5	5	5	5	5	5	5	5	5
	LEOPARD	1	10%	16	16	16	16	16	16	16	16	16	16
	LEOSAR	1	20%	31	31	31	103	31	31	31	31	31	31
on during transfer	LEVIS	1	10%	4	4	4	4	4	4	4	4	4	4
	LunarMag: Daughter	0	5%	0	0	0	0	0	0	0	0	0	0
	MIMO	1	30%	54	54	54	54	54	54	54	54	54	54
	PRARE-L: Daughter	0	20%	0	0	0	0	0	0	0	0	0	0
on during transfer	RadMo	1	20%	11	11	11	11	11	11	11	11	11	11
	RaPS: Daughter	0	20%	0	0	0	0	0	0	0	0	0	0
	SERTIS	1	10%	15	15	15	15	15	15	15	15	15	15
	SLR	0	30%	0	0	0	0	0	0	0	0	0	0
	SPOSH	1	20%	3	3	10	10	10	3	3	10	3	10
	USMI	1	30%	10	10	10	10	10	34	10	10	10	10
	VIS-NIR	1	20%	5	5	5	5	5	15	5	5	5	5
	XRF-L	1	30%	30	30	9	9	9	30	30	9	30	9
Thermal				**50,0**	**50,0**	**50,0**	**50,0**	**50,0**	**50,0**	**50,0**	**50,0**	**50,0**	**50,0**
PDHT				**224,0**	**224,0**	**224,0**	**224,0**	**224,0**	**224,0**	**224,0**	**224,0**	**224,0**	**224,0**
	PMM	1	10%	50,0	50,0	50,0	50,0	50,0	50,0	50,0	50,0	50,0	50,0
	Modulator	1	20%	10,0	10,0	10,0	10,0	10,0	10,0	10,0	10,0	10,0	10,0
	TWTA	1	20%	141,0	141,0	141,0	141,0	141,0	141,0	141,0	141,0	141,0	141,0
	HGA Pointing Mechanism	1	20%	23,0	23,0	23,0	23,0	23,0	23,0	23,0	23,0	23,0	23,0
PMC				**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**	**20,0**
	PMC	1	5%	20,0	20,0	20,0	20,0	20,0	20,0	20,0	20,0	20,0	20,0
Power				**53,8**	**45,6**	**44,8**	**48,3**	**44,4**	**47,0**	**45,2**	**44,4**	**45,2**	**44,4**
	PPDU	1	10%	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0	15,0
	Conversion loss 50V-->28V	0,0%	5%	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0	0,0
	Harness Loss	3,5%	5%	24,7	19,5	19,0	21,2	18,7	20,4	19,2	18,7	19,2	18,7
	Distribution Loss	2,0%	5%	14,1	11,1	10,8	12,1	10,7	11,6	11,0	10,7	11,0	10,7
Total				**760**	**601**	**587**	**655**	**579**	**629**	**593**	**579**	**593**	**579**
Margin													
Units margin				134	104	99	112	98	112	103	98	103	98
System margin			10%	89	71	69	77	68	74	70	68	70	68
PSS total with margin	W			**983**	**776**	**754**	**844**	**744**	**815**	**765**	**744**	**765**	**744**

The Ka-Band link budget is given in the table below. It respects the required link margin of 3dB.

Table 4.5-11: Ka-band link budget

DOWNLINK BUDGET		
LEO Ka-Band Link Budget Weilheim 09		
QPSK		
LINK PARAMETERS	**UNIT**	**Nominal**
Frequency	**MHz**	26000
TX Power	**dBm**	48,00
Circuit losses from TX to antenna	**dB**	1,50
S/C TX Ant. gain	**dBi**	43,82
Satellite EIRP	**dBWi**	60,29
GS elevation angle	**°**	10,00
SC elevation angle	**°**	0,88
Slant range	**km**	405543,97
Free space losses	**dB**	232,91
Atm. & Rain Attenuation	**dB**	3,32
Propagation loss	**dB**	236,41
G/S Station G/T	**dB/°K**	41,98
DOWNLINK C/No	**dBHz**	93,96
TELEMETRY RECOVERY		
Implementation loss (Mod/Demod)	**dB**	3,00
TM Bit Rate (incl.form. excl.coding)	**Mbps**	204
Coding		RS255/239 + conv. 3/4
Required BER		1,00E-09
Required Eb/No	**dB**	4,00
TM recovery Margin	**dB**	3,86

PSS to Platform Interfaces

Mechanical Interfaces

The interfaces to the platform, sub-satellites and solar array are indicated in Figure 4.5-24 and Figure 4.5-25. The interface points of the sub-satellites are of interest for the platform since it has to provide hard points for a proper load transfer. Concerning the solar array, mainly the position of the solar array drive is important for the platform since all loads of the deployed solar array are transferred via this item.

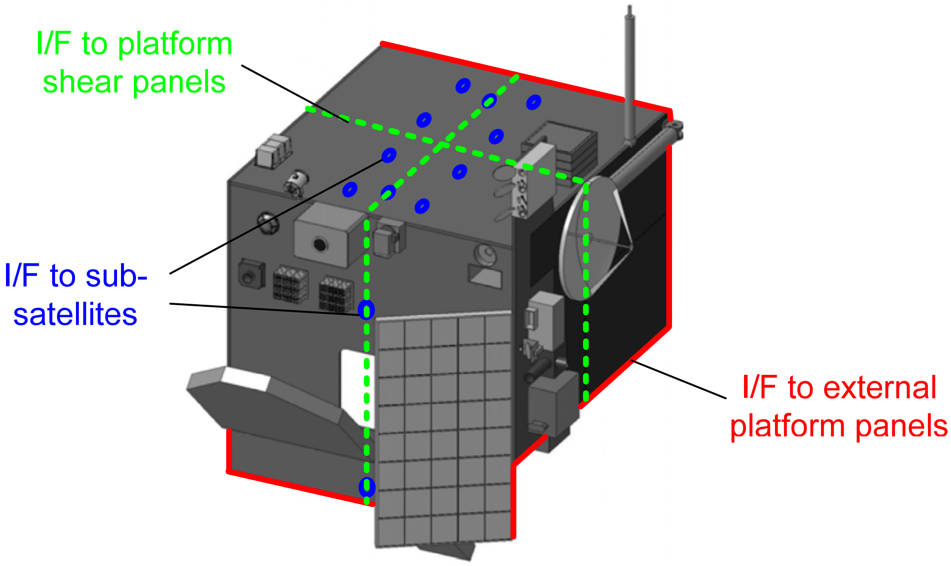


Figure 4.5-24: Iso view of payload support system

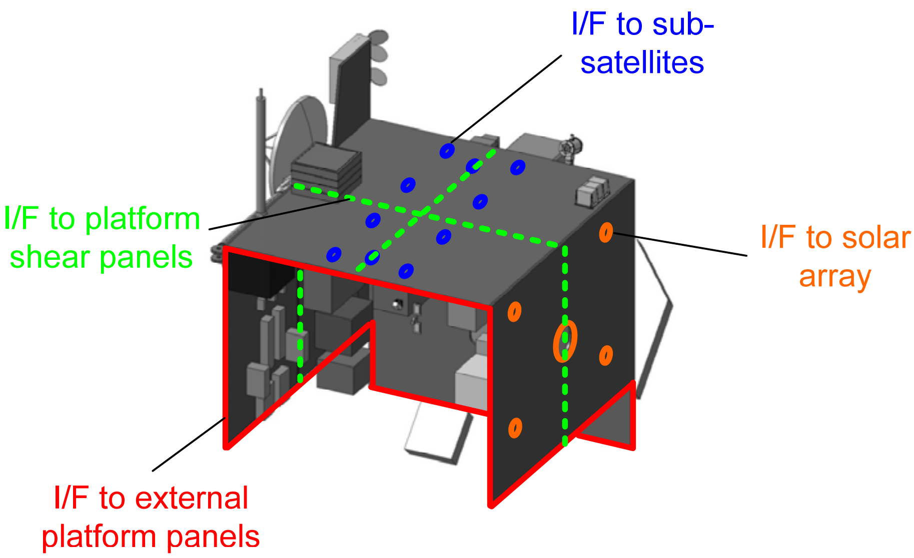


Figure 4.5-25: Iso view of payload support system

Thermal Interfaces

The payload support system is thermally fully coupled with the platform. Hence, its thermal properties are to be iterated in the frame of the platform thermal analysis (including payload support system).

For the externally mounted payloads it is required that there is no heat transfer to the payload support system and the platform. The allowed residual thermal fluxes are to be specified.

Electrical interfaces

The electrical interfaces between platform and payload section are kept to a minimum in order to reduce the interface and operational complexity. They comprise the following:

- Power supply 28V regulated (50V to 28V converter allocated in platform)
- DNEL line (disable non essential loads) generated from the platform in case of battery undervoltage
- Release actuators on payload side with pulse generator in the platform EPS
- Interprocessor link (Milbus) between platform OBDH and PMC
- TBD other direct TC from platform to the payload (mainly PMC and PPDU commanding)
- Direct discrete TM from the payload to the platform like on/off status and dedicated thermistors
- Separation TM of the subsatellites from the separation bridge
- X-band RF interface between platform TT&C and payload HGA
    - Type: Waveguide interface
    - HGA Cassegrain antenna diameter: 0.75m
    - X-band antenna gain @ 7213MHz / TC: 32.7dBi
    - X-band antenna gain @ 8475MHz / TM: 34.09dBi
    - X-band TM & TC rotary joint losses (2 joints): 0.2dB
    - Line losses depend on the boom length and remaining cable or waveguide length to the TT&C transponder.
    - HGA pointing loss: neglegible

4.5.2.4 Main Satellite Budgets

Margin Philosophy

The OHB margin philosophy shown in the following sections is in-line with the GDIR requirements [AD4].

Mass Margin

Table 4.5-12 lists the basis, how the mass margins are defined in the LEO study.

Table 4.5-12 Mass Margin Philosophy

Classification	Category	Assigned Margin
Existing hardware	A	5%
Design based on existing hardware requiring minor modification	B	10%
Detailed design or design based on exiting H/W requiring major modification	C	15%
Preliminary design or equipment not yet developed	D	20%

The mass budget has been established with a bottom-up approach where the unit masses were either estimated bottom-up from PCB level e.g. for avionic units or by using standard values for the mass per area or per length e.g. for MLI blankets. Furthermore, responses from possible suppliers to Requests For Information (RFI) issued by OHB-System or other projects were used.

On System level, a System Mass Margin of 15% is required – in answer to the recent mass increase according to the final SFS **a system margin of 10.8% was assumed to meet the Soyuz Fregat Launch performance.** This measure shows the actual need for mass reduction of minimum 4.2% on dry mass.

Power Margin

In the current phase of the project, the power budget is a tool to support the sizing of power subsystem which will be adapted accordingly.

As the payload power range for the LEO mission is expected to have the largest impact on the power subsystem, the sizing of the power subsystem will be revisited after final instrument resource definition.

According to [AD4], maturity margins are assigned to the units of the power system. These margins are listed in Table 4.5-13.

Table 4.5-13 Design maturity margins for power

ECSS Classification	Category	Assigned Margin
Off-the-shelf equipment requiring no modification.	A/B	5%
Off-the-shelf equipment requiring minor design modifications.	C	10%
Newly designed and developed equipment or existing equipment requiring major redesign.	D	20%
Preliminary design or equipment not yet developed	D	30%

Mass Budget

Table 4.5-14 shows the Dry Mass Budget for the LEO satellite platform for injection into LTO. The subsatellite mass is taken according to the mass apportionment for them with 110kg plus 20% system margin.

Table 4.5-14 LEO Dry Mass Budget for the LTO Option

Subsystem	No. Units	Unit Mass (kg)	Mass excl. Margin (kg)	Component Margin (%)	Component Margin (kg)	Mass inc. Comp. Margin (kg)
Scientific Payload (Main Orbiter)			**125.4**	**20**	**25.0**	**150.4**
ATON	1	1.0	1.0	20	0.2	1.2
HRSC-L	1	25.0	25.0	20	5.0	30.0
LEOPARD - Langmuir Probe	1	0.9	0.9	10	0.1	1.0
LEOPARD - Dustdetector	1	6.5	6.5	20	1.3	7.8
LEOSAR	1	20.5	20.5	20	4.1	24.6
LEVIS	1	2.1	2.1	20	0.4	2.5
MIMO	1	26.4	26.4	20	5.3	31.7
RadMo	1	7.4	7.4	20	1.5	8.9
SERTIS	1	3.5	3.5	20	0.7	4.2
SPOSH	1	2.5	2.5	20	0.5	3.0
USMI	1	7.0	7.0	20	1.4	8.4
VIS-NIR	1	10.0	10.0	20	2.0	12.0
XRF-L	1	12.6	12.6	20	2.5	15.1
Payload Support S/S			85.8	15	12.7	98.5
Payload Power Distribution Unit (PPDU)	1	11.2	11.2	10	1.1	12.3
Payload Management Computer (PMC)	1	8.4	8.4	5	0.4	8.8
Data Mass Memory (PMM)	1	16.5	16.5	10	1.7	18.2
Scientific Data Link Ka-Band (PDT)	1	46.3	49.7	19	9.5	59.2
Platform (Main Orbiter)			**540.1**	**12**	**64.9**	**605.0**
TM/TC S/S			8.6	5	0.4	9.0
Low Gain Antennas X-Band	2	0.2	0.3	5	0.0	0.3
X-Band Transceiver	2	3.6	7.2	5	0.4	7.6
Waveguide Switch	1	0.2	0.2	5	0.0	0.2
Combiner & Hybrid	3	0.3	0.9	5	0.0	0.9
AOCS			48.4	6	2.7	51.0
Star Tracker Unit incl. Electronics	2	2.6	5.2	10	0.5	5.7
Inertial Measurement Unit	1	8.4	8.4	5	0.4	8.8
Reaction Wheel	4	8.5	34.0	5	1.7	35.7
Coarse Sun Sensors	24	0.0	0.8	5	0.0	0.8
OBDH S/S			15.7	10	1.6	17.3
DMU	1	15.7	15.7	10	1.6	17.3
Electrical Power S/S			95.4	8	7.7	103.1
Solar Panels (incl. harness, joints,yoke)	2	18.0	36.0	5	1.8	37.8
Solar Array Drive Electronic	1	2.5	2.5	10	0.3	2.8
Battery Unit	2	14.5	29.0	10	2.9	31.9
Power Conditioning Unit (PCU)	1	16.9	16.9	10	1.7	18.6
Power Distribution Unit (PDU)	1	11.0	11.0	10	1.1	12.1
Harness			39.0	20	7.8	46.8
PPS Harness	1	5.0	5.0	20	1.0	6.0
TM/TC Harness	1	1.5	1.5	20	0.3	1.8
Power Harness	1	13.0	13.0	20	2.6	15.6

Subsystem	No. Units	Unit Mass (kg)	Mass excl. Margin (kg)	Component Margin (%)	Component Margin (kg)	Mass inc. Comp. Margin (kg)
Data Harness	1	19.5	19.5	20	3.9	23.4
Propulsion S/S			104.6	9	9.9	114.4
He Supply Assembly	1	25.5	25.5	9	2.3	27.8
MON Supply Assembly	1	30.9	30.9	9	2.9	33.8
MMH Supply Assembly	1	30.9	30.9	9	2.9	33.8
10 N Thruster	12	0.7	7.8	5	0.4	8.2
400 N Thruster (LAE)	1	3.5	3.5	5	0.2	3.7
Piping & Mounting	1	6.0	6.0	20	1.2	7.2
Structure S/S & Mechanisms			187.6	15	28.5	216.1
Primary Structure			138.2	15	20.7	158.9
Secondary Structure			35.5	15	5.4	40.9
Mechanism			13.8	17	2.3	16.2
Thermal Control S/S			40.9	15	6.3	47.2
External MLI	1	10.0	10.0	10	1.0	11.0
Internal MLI	1	2.0	2.0	10	0.2	2.2
High Temperature MLI	1	0.4	0.4	10	0.0	0.4
Paint/ Surface	1	10.5	10.5	20	2.1	12.6
Heat Pipes	1	2.5	2.5	10	0.2	2.7
Heaters	1	3.0	3.0	10	0.3	3.3
Temperature Sensors	1	0.5	0.5	10	0.0	0.5
Temperature Control	1	0.3	0.3	10	0.0	0.3
Interface Filler	1	0.8	0.8	20	0.2	1.0
Miscellaneous (Washers, I/F Filler, Adhesive, etc.)	1	1.0	1.0	20	0.2	1.2
Main Radiator Baffle	1	10.0	10.0	20	2.0	12.0
LEO Orbiter Dry Mass excl. SubSatellite			**751.29**	**13.65**	**102.56**	**853.86**
				Systemmargin:		
LEO Orbiter Dry Mass incl. System Margin			**853.86**	**10.8%**	**92.22**	**946.07**
SubSatellite incl. Payload & Propellant			**255.8**	**3%**	**7.3**	**263.1**
SubSatellite 1	1	120.9	120.9	3%	3.6	124.5
SubSatellite 2	1	120.9	120.9	3%	3.6	124.5
SubSatellite Propellant	2	7.0	14.0			14.0
LEO Orbiter Dry Mass incl. SubSatellite			**1109.66**	**8.96%**	**99.47**	**1209.13**

Propellant Budget

Table 4.5-15 provide the ΔV and propellant mass budget for the baseline launch scenario with LTO injection and subsequent lunar orbit insertion by means of the CPPS. ΔV figures are taken from Chapter 4.3.3.

Table 4.5-15 LEO Delta-V and Chemical Propulsion Mass Budget for the LTO Case

	Nominal Delta V [m/s]	Margin [%]	Margin [m/s]	Delta V inc. Margin [m/s]	Specific Impulse [s]	Propellant Mass [kg]	Orbiter Mass [kg]
Orbiter Drymass inc. Systemmargin (excl. Subsatellite)							**946.1**
Pressurant						**5.0**	
Helium Pressurant						5.0	951.1
Propellant Residual						**15.0**	
Propellant Residual (≤2%)						15.0	966.1
Propellant Reserve						**0.0**	
Propellant Reserve						0.0	966.1
Moon Orbit 90°	**300.0**	**10.0**	**30.0**	**330.0**		**127.4**	
Orbit Maintenance 1 year	300.0	10	30.0	330.0	271.5	127.4	1093.5
Moon Orbit 85°	**211.0**	**36.0**	**76.0**	**287.0**		**124.4**	
Inclination Change	150.0	10	15.0	165.0	271.5	69.9	1163.4
Orbit Maintenance 3 years	61.0	100	61.0	122.0	271.5	54.5	1217.9
Subsatellite Separation							
Subsatellite mass inc. Systemmargin							**263.1**
Subsatellite Separation							1481.0
Moon Orbit Acquisition	23.0	10.0	2.3	25.3		14.1	
100km Moon Orbit Acquisition	23.0	10	2.3	25.3	271.5	14.1	1495.1
Moontransfer	855.0	13.3	114.0	969.0		543.6	
Moon Injection Manoeuvre	780.0	5	39.0	819.0	318.5	448.1	1943.2
LTO Corrections	75.0	100	75.0	150.0	318.5	95.6	2038.7
Total Delta V	**1389.0**	**16.0**	**222.3**	**1611.3**			
Orbiter Propellant Mass						**824.6**	
Orbiter Launch Mass (inc. Subsatellite)							**2038.7**

Power Budget

The Main Satellite's power budget combines the power requirements of the Payload Support System and the Platform, as these need to be considered together for the sizing of the Solar Array and the battery.

Preliminary Linkbudgets

Preliminary Budget for HGA Usage on Moon

Downlink

Based on the assumptions from the GSOC for the 16 m ground station a preliminary link budget has been established.

Table 4.5-16 Preliminary Downlink Budget for HGA

Satellite Parameters			
Link Distance =	406100		km
Antenna Diameter =	0.7		m
Frequency Band =	X-Band		
Bitrate =		0,004	Mbit/s
Downlink Parameter			
Frequency =		8400	MHz
Bitrate=		4000	bps
Modulation =			QPSK
Coderate=		0,5	
required BER	1,0E-06		
Erford. Eb/N0 =	4,5		dB
Transmit Power	0,25		W
Transmit Antenna Gain =	32,7		dBi
Polarisation loss =	0,3		dB
Pointing loss =	0,5		dB
Miscellaneous losses=	4		dB
Implementation loss	2		dB
EIRP=		22,68	dBW
Free Space Loss =	223,10		dB
Transmission losses =	227,90		dB
G/T Receiver =	36,6		dB/K
C/No =		63,98	dBHz
Required C/No =	45,53		dBHz
Margin =		**18,45**	**dB**
Power Density	-46,81		dBW/Hz
Transmit Power in dB/mW	23,98		dBm

Uplink

Based on the assumptions from the GSOC for the 16 m ground station a preliminary link budget has been established.

Table 4.5-17 Preliminary Uplink Budget for HGA

Satellite Parameters		
Link Distance =	406100	km
Antenna Diameter =	0.7	m
Frequency Band =	X-Band	
Bitrate =	0,001	Mbit/s
Downlink Parameter		
Frequency =	7200	MHz
Bitrate=	1000	bps
Modulation =		QPSK
Coderate=	1	
required BER	1,0E-06	
Erford. Eb/N0 =	10,5	dB
Transmit Power	10	W
Transmit Antenna Gain =	61	dBi
Polarisationloss =	0,3	dB
Pointing loss =	0,5	dB
Miscellaneous losses=	4	dB
Implementation loss	2	dB
EIRP=	67,00	dBW
Free Space Loss =	221,76	dB
Transmission losses =	226,56	dB
G/T Receiver =	-3,9	dB/K
C/No =	69,14	dBHz
Required C/No =	42,50	dBHz
Margin =	**26,64**	**dB**
Power Density	-24,77	dBW/Hz
Transmitt Power in dB/mW	40,00	dBm

4.5.3 Sub-Satellites

4.5.3.1 Configuration

The overall configuration of the LEO subsatellite is depicted in Figure 4.5-26.

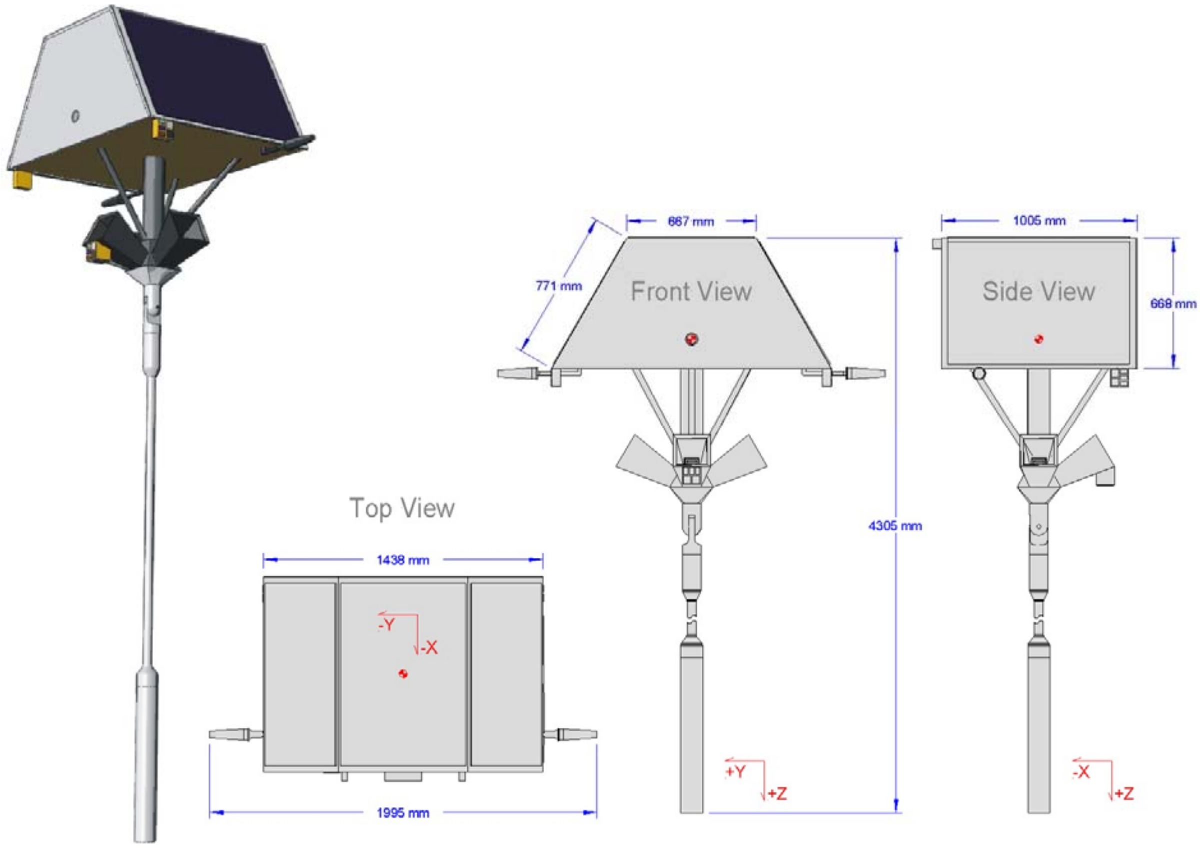


Figure 4.5-26: LEO Subsatellite Overall Configuration

The X-sided closure area of the subsatellite main body is of trapezoidal shape with the main dimensions indicated (base: 1438 mm, roof: 667 mm; legs: 668 mm). The main body itself consists of a cube shaped body which houses the majority of the electronic equipment. The outside of this cube is used for the accommodation of the solar generator roof, the RF antennas as well as 4 of the six RaPS heads (-X, +Z, ± Y).

On the lower side of the main body cube the boom root is mounted. The root provides the space for the accommodation of the three star sensor heads and the hinge for the deployable boom. The end piece of the deployable part of the boom is thickened to allow the attachment of the magnetometer sensor head assembly in form of an 800 mm cylinder.

The design concept leads to an inherently gravity gradient stable satellite which allows minimizing the effort on the attitude control. The accommodation concept within the main body is depicted in Figure 4.5-27. It shall be noted that the +X direction is defined as the flight direction of the trailing subsatellite and is referred to as the front side (front panel) of the spacecraft. The +Z direction refers to zenith orientation in the nominal flight orientation and the Y direction completing a right handed orthogonal system.

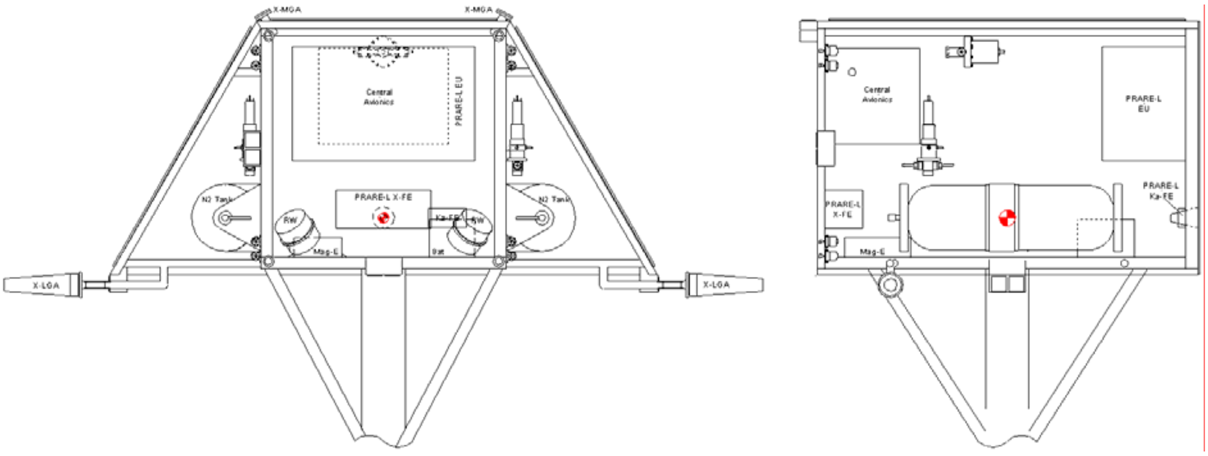


Figure 4.5-27: LEO Main Body Accommodation Overview

On the inner side of the front panel the PRARE-L electronics unit (EU), the Ka-Band horn antenna and the Ka-Band Front-End (Ka-FE) equipment will be accommodated. The central avionics as well as the X-Band portion of the PRARE-L, i.e. the X-band Front-End, the medium gain antennas (MGA) as well as the interconnecting harness will be mounted on the inner side of the rear panel. The low gain X-band antennas are mounted on short stubs to each side of the spacecraft to allow for a free hemispheric view for each antenna.

The bottom plate of the main body houses magnetometer electronics, the star camera electronics, the battery and the four reaction wheels. The propulsion tanks, the major portions of the propulsion pipe work and the thrusters are mounted on the outside surfaces of the main body cube. The pressure regulator is mounted on the inner side of the main body top panel.

The arrangement of all units is balanced such that the centre of mass (Figure 4.5-27) is in the middle between the two symmetrically located propulsion tanks and at the same vertical (Z) position as the phase centre of the Ka-band horn antenna. With this location of the centre of mass it is ensured that

- the centre of mass, as required by the science application, is on the elongation of the connecting line between the two spacecraft Ka-band horn antenna phase centres,
- the centre of mass does not change over the mission life time due to the even depletion (at same temperature condition) of the two propulsion tanks.

4.5.3.2 Payload Accommodation

The payload to be accommodated on the LEO subsatellite consists of

- the LunarMag instrument (red colour in Figure 4.5-28)
  - the LunarMag electronics unit
  - two LunarMag sensor bracket including two sensor heads
- the PRARE-L instrument (green colour in Figure 4.5-28, consisting of
  - the PRARE-L EU (Electronic Unit))
  - the PRARE-L Ka-FE (Front-End)
  - the PRARE-L Ka-Band Horn Antenna
  - the PRARE-L X-FE

– two PRARE-L X-Band Antennas
- the RaPS instrument (blue colour in Figure 4.5-28) consisting of the 6 heads.

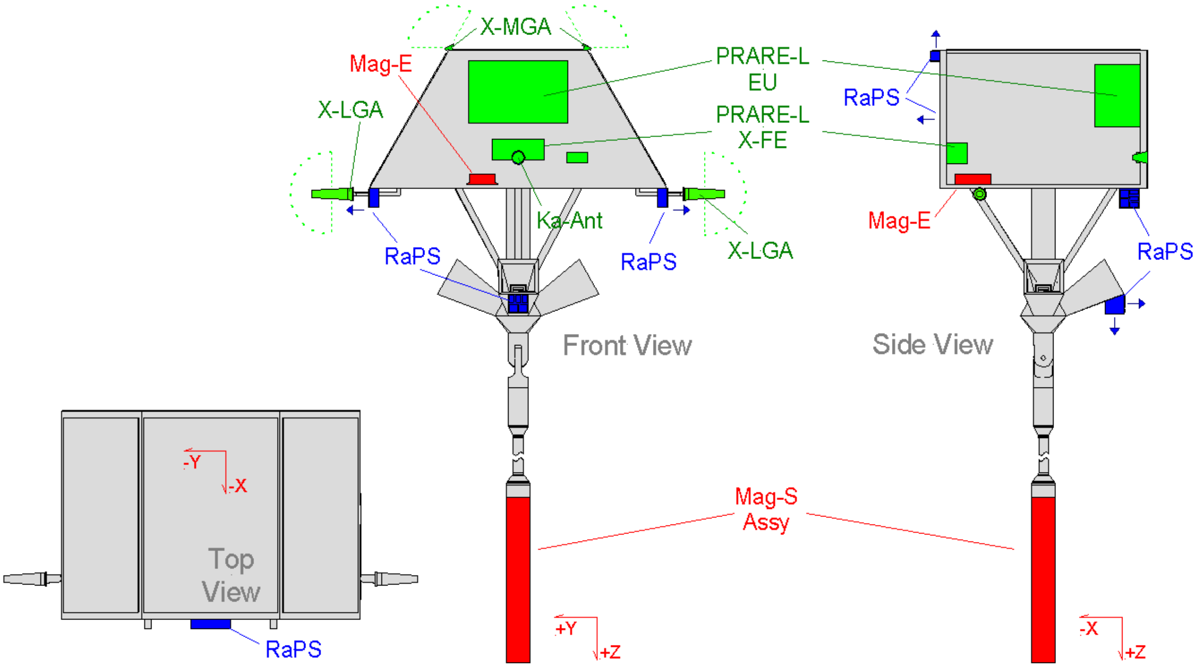


Figure 4.5-28: Subsatellite Instrument Accommodation Overview

The LunarMag electronics is accommodated on the bottom plate of the satellites main structure. The two sensors are accommodated within a cylindrical sensor assembly that is pre-integrated and tested by the scientists.

The RaPS sensor heads are accommodated on the lower side of the spacecraft solar array panels (±Y), on the -X panel (+Z), at the lower side of the +X star sensor (+X, -Z) and on the -X panel (-X).

4.5.3.3 Electrical Architecture

The electrical architecture of the subsatellites as depicted in Figure 4.5-29 represents a typical small satellite architecture. The main element within this concept is the central avionics which concentrates all data handling and power control/distribution function and as such provides all necessary power and data resources to the spacecraft and its users.

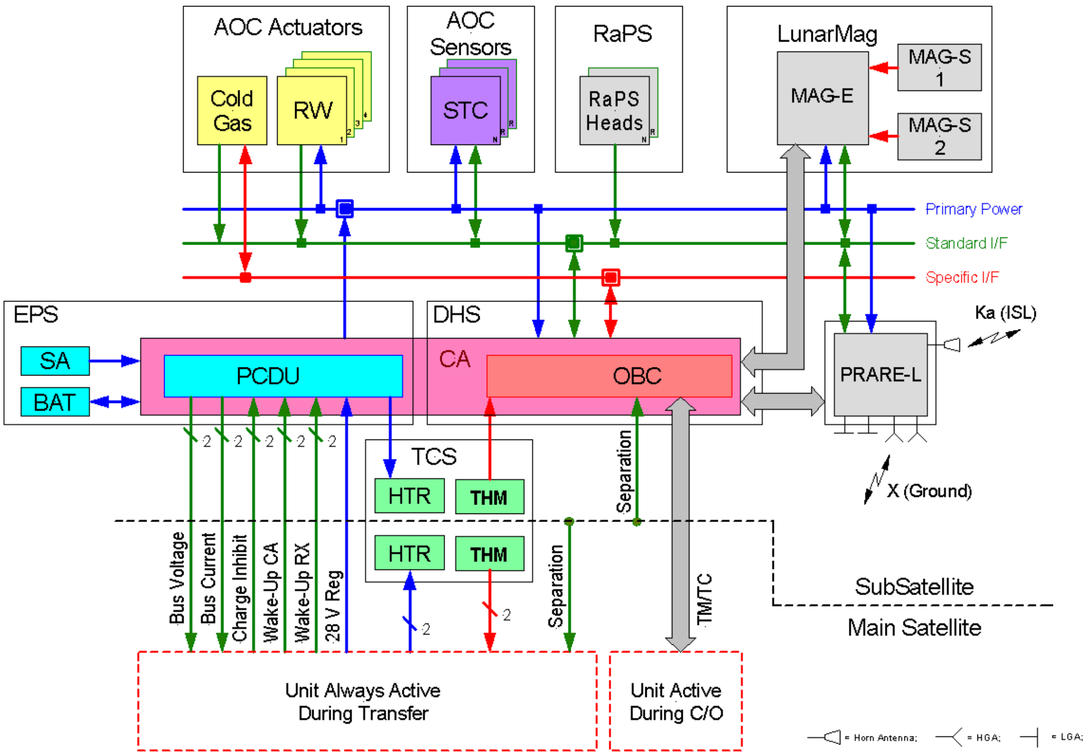


Figure 4.5-29: Subsatellite Electrical Architecture and Main Satellite Interfaces

Data Handling

The on-board data handling constitutes the heart of the satellite data management and handling system and integrates all relevant functions of the spacecraft bus. This function is in particular responsible for the

- reception of telecommand data stream (base-band level) from the X-band receiver
- decoding of hardware decodable commands (high priority commands) from the tele-command data stream, its execution by-passing all software resources as such providing direct ground access to essential spacecraft functions
- decoding of nominal bus commands using the processor software and issuing of the respective commands to all on-board (bus and payload) users
- decoding, intermediate storage and transmission of time tagged commands and execution at the foreseen time events, both for bus and instrument
- acquisition, time-stamping, intermediate storage of bus and payload relevant housekeeping and ancillary data, adequate to monitor the status and performance of the entire spacecraft and its subsystems
- transmission of stored and real-time housekeeping data to the X-band transmitter during periods of contact with the ground station
- provision of interfaces to acquire and condition housekeeping and attitude control sensor data
- provision of interfaces to command on-board equipment, in particular attitude and orbit control actuators
- provision of fault detection and reporting as well as autonomous execution of pre-defined on-board isolation and recovery measures (FDIR)
- provision of adequate computing and memory resources for the different portions of operating and application software that is necessary to operate the spacecraft autonomously

during periods without ground contact, in particular the

- execution of the attitude and orbit control system algorithms
- supervision and management of all on-board resources
- execution of the thermal control functions for bus and instrument
- execution of fault detection, reporting and autonomous on-board isolation and recovery to the extent defined for each mission
- commanding, control and supervision of payload / instrument.

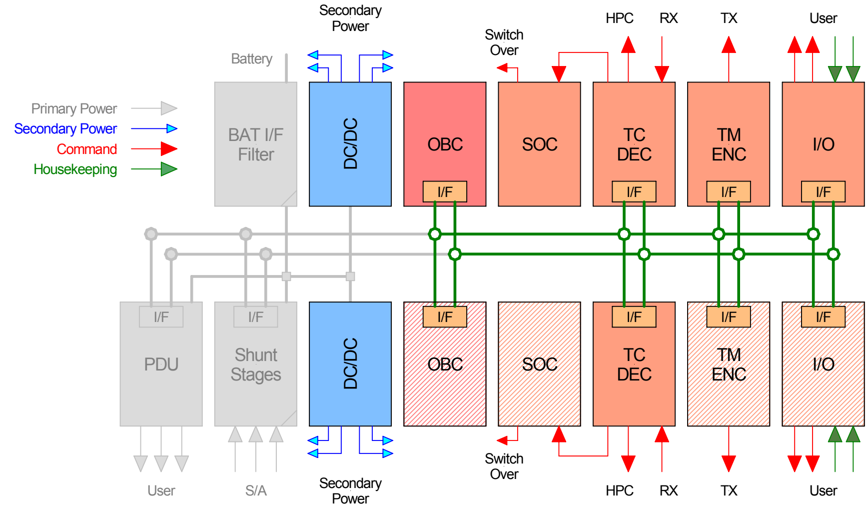


Figure 4.5-30: Data Handling Portions within the Central Avionics

The data handling functions are implemented in the respective portions of the central avionics that additionally includes power control and distribution functions, as well. The major concept of the central avionics architecture w.r.t. data handling is summarized in Figure 4.5-30. Besides the secondary power supply which is shared with the power functions the data handling consists of the following modules:

- the central on-board processor unit (OBC),
- the switch-over controller (SOC),
- the telecommand decoder (TM DEC),
- the telemetry encoder (TM ENC) including the on-board memory,
- the input / output modules (I/O).

The data flow on the subsatellites is schematically summarized in Figure 4.5-31 Uplink data will either be received via the medium gain antennas (nominal conditions) or via the low gain antennas (off-nominal conditions). The PRARE-L internal receiver will demodulate the data from the carrier and forward the command data to the central avionics. The data

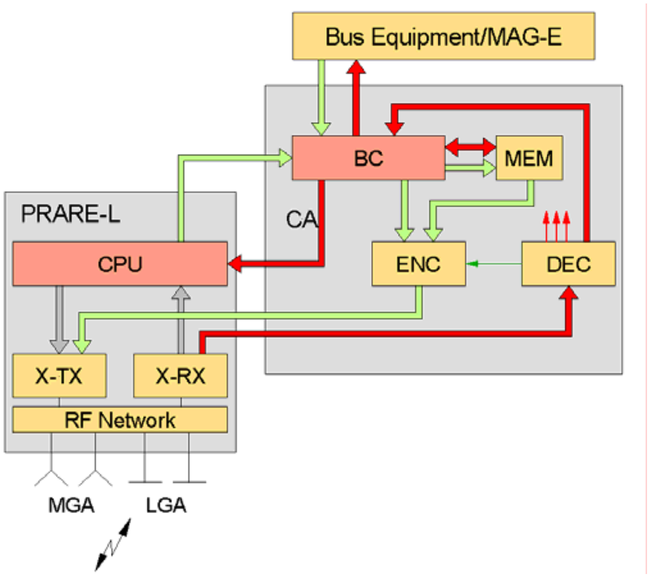


Figure 4.5-31: Subsatellite Data Flow Concept

stream will be decoded by the central avionics internal CCSDS compatible decoder which separates the direct commands from the derail data stream dedicated for the board computer execution. In the other direction, real-time and memorized data will be encoded by the central avionics internal encoder and routed to the PRARE-L transmitter for modulation of the downlink RF signals. Depending on the operational state, the RF signals will be routed via a medium or high gain antenna to Earth.

Electrical Power

The electrical power system's main tasks are to

- route during sun lit periods with high efficiency power from the solar generator to the internal main bus and further on to on-board users and the battery
- establish a safe battery charge control
- receive from the on-board data handling switching commands for the on-board electrical users and heaters
- receive from the on-board data handling redundancy switch-over commands for cold redundant internal portions
- acquire and transmit power system relevant housekeeping data to the on-board data handling system to allow the on-board control and the ground to adequately evaluate the health status of the entire power system
- as a response to switching commands received from the on-board data handling, distribute via over-current protected outlets unregulated / adequately regulated voltage to all on-board users
- activate ordnance currents for deployment mechanisms upon receipt of protected commands from the on-board data handling
- receive regulated electrical power from the main spacecraft during coupled transfer and check-out periods.

The power system consists of three solar generators which are located on the top and side panels of the spacecraft, the battery (BAT) and the power control and distribution module (PCD) located within the core avionics. The PCD itself consists of the solar array regulator realised as shunt stages, the bus filter and battery interface module, the power distribution module (PDU) and the DC/DC converters which are shared with the data handling modules. The control software for battery charging is embedded in the on-board computer as an application software package.

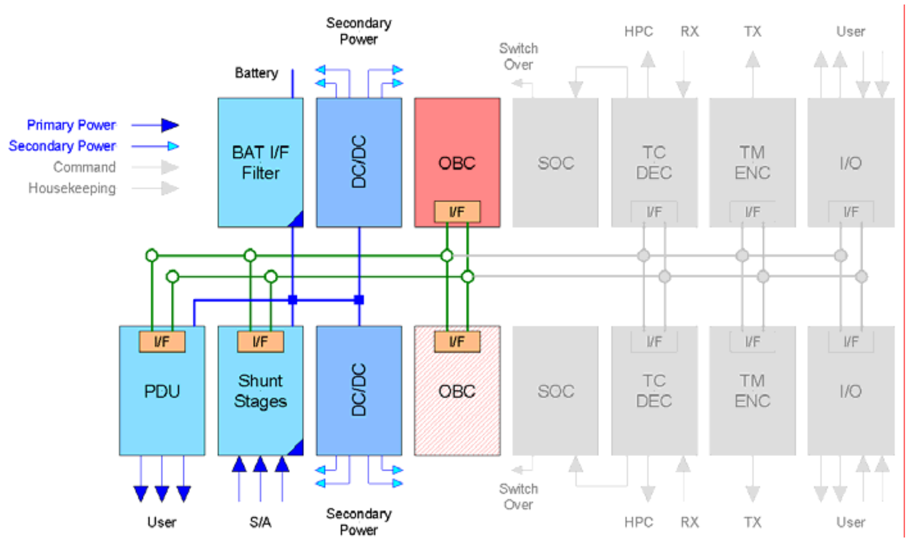


Figure 4.5-32: Power Control & Distribution Portions inside the Central Avionics

The reference battery foreseen for the LEO application is a recurring unit available on the market and already used for a magnetically demanding mission. The battery consists of 16 parallel strings with 8 cells in series, referred to as 8s16p type battery. With a cell capacity of 1.5 Ah the total battery capacity adds up to 24 Ah at BOL conditions.

The solar array is composed of three solar generator panels that are located on the top (-Z) and the side (-Y/-Z, +Y,-Z) panels. Each side panel consists of 11 strings with 18 GaAs Triple Junction (TJ) cells each and the top panel of 10 strings with 18 GaAs TJ cells each. The solar generator is capable to provide an average bus power of 100 W (150 W) on a typical noon orbit (dawn-dusk orbit).

RF Communications

The overall concept of the RF communication system of the subsatellite is schematically outlined in Figure 4.5-33. The receiver and transmitter functions are part of the PRARE-L instrument, the interconnection RF hardware part of the satellite bus. The depicted scheme allows routing the signals from either the medium gain antennas (MGA) or the low gain antennas (LGA) to both of the active receivers. The active transmitter can be routed to each of the four antennas. The demodulated received signals will be transferred to the central avionics for further processing while the downlink data will be accepted by the PRARE-L transmitter for the modulation of the downward RF link.

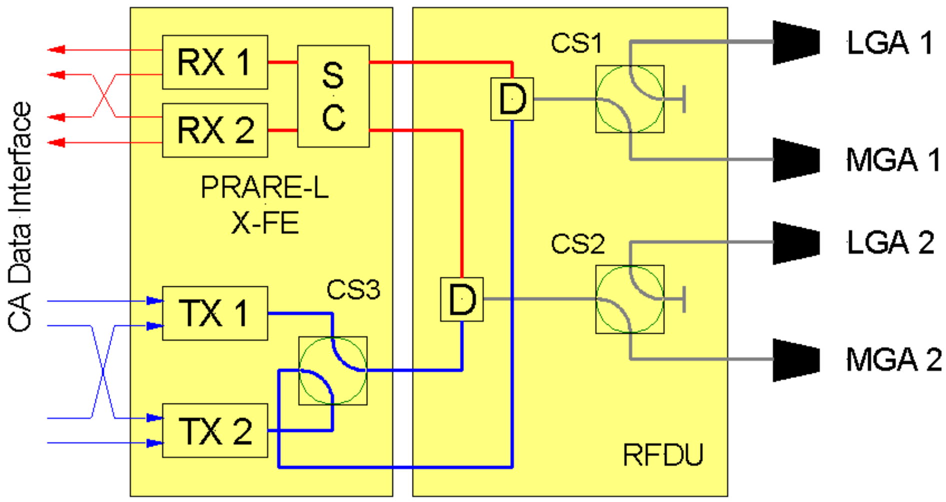


Figure 4.5-33: RF Communication Architecture

4.5.3.4 Structural Design

The mechanical composition of the subsatellite structure and the description of its main elements are given in Figure 4.5-34. The main body (1, 2) will be manufactured from carbon reinforced plastic (CRP) based sandwich panels with a honeycomb made from Aluminium. The front side will be amended with a trapezoid, metallic surface that will provide an undisturbed surface for the Ka-band horn antenna. The solar generator (3) consists of three panels in form of a roof, two panels on the side which are suspended with the support struts and one on the top that is directly mounted to the main body. The root boom (4) and the deployable boom (5) are manufactured from carbon fibre reinforced plastics and provide at the end the attachment space for the magnetometer sensor assembly.

The overall arrangement of the structural elements is such that the flow of integration and test is eased to an utmost extent as

- all RF equipment and the interconnecting harness are mounted on an U-shaped structure together with all major electronics elements such that electrical testing can be done nearly exclusively on this panel,
- all propulsion equipment and interconnecting pipe work are mounted on a reverse U-shaped structure such that all propulsion related integration and pre-testing can be performed on this level,
- both U-shaped structures allow (with an appropriate support structure) for good access to all equipment during the integration process,
- the solar generator and the magnetic sensor assembly can be attached at the very last steps of integration and test,
- the boom deployment mechanism can be tested alongside all electrical tests and may be integrated in a very late stage of the overall handling flow.

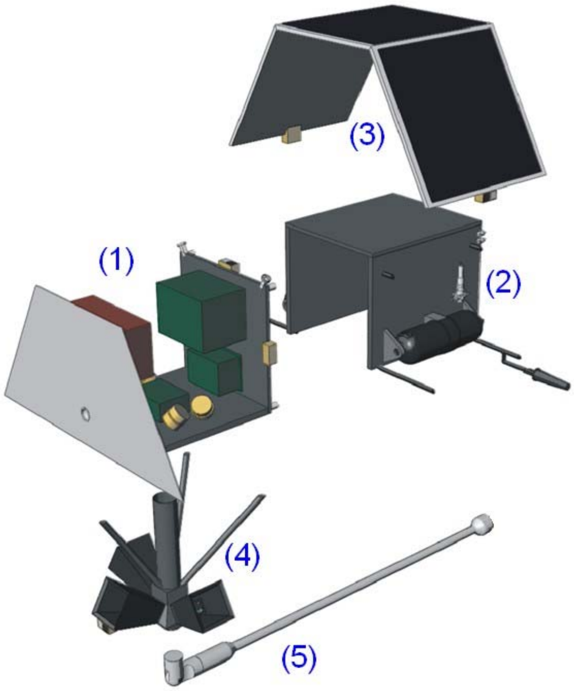


Figure 4.5-34: Subsatellite Structural Concept

4.5.3.5 Propulsion

The main task of the propulsion system for the LEO mission is to desaturate on demand the reaction wheels of the attitude control system and, whenever needed, execute the required orbit manoeuvres. Due to the benign requirements concerning the attitude and orbit control manoeuvres and in order to establish a platform optimized for gravity measurements the propulsion system has been based on the use of cold gas as its propellant. The schematic concept of the established system is depicted in Figure 4.5-35.

The necessary cold gas supply is stored under high pressure in two tanks that are symmetrically arranged around the spacecraft's centre of mass. The pressure in this regime will be measured by a dedicated high pressure gauge and its values regularly acquired by the central avionics. Propellant can be added and relieved from the system via a fill / drain valve during ground operations. Via the pressure regulator (PR) the pressure will be reduced to fit for the operation of the thrusters. The pressure regime after the regulator will be monitored using a dedicated low pressure gauge. Any unwanted over-pressure in this regime will lead to the opening of the relief valve.

The low pressure regime consists of two thrusters branches each consisting of 4 thrusters. In cases of thruster leakage the affected branch can be shut down with a latch valve such that the remaining branch is protected.

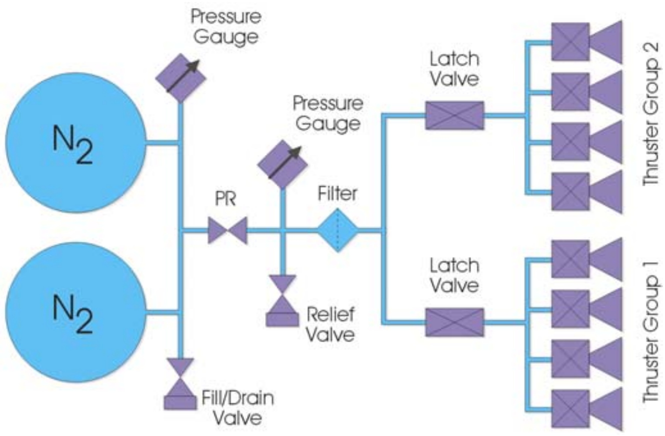


Figure 4.5-35: Subsatellite Propulsion Concept

4.5.3.6 AOCS

The main functions of the attitude and orbit control for the subsatellites are

- Determination of the satellite's attitude using appropriate sensors
- Control the satellite's attitude with predefined accuracy using adequate actuators
- Determination of the spacecraft position and velocity or, equivalently, its orbital elements as a function of time based on an orbit propagator, regularly updated by the ground (resulting on PRARE-L tracking measurements and refinements in the gravity model)
- Execution of orbit control using the satellite's propulsion subsystem, e.g. adjusting the orbit to meet predetermined conditions including:
  - Initial set-up of the operational flight constellation
  - Correction manoeuvres, if necessary, depending on the final stability of the selected target moon orbit
  - Controlled impact on the moon surface at the end of mission, control accuracy as the remaining fuel allows
- Acquisition and transmission of attitude / orbit determination and control dedicated equipment's housekeeping data (health and status) to the on-board data handling system

The driving requirements for the attitude control result from the pointing of the two PRARE-L Ka-band antennas towards each other during range rate measurements via the satellite-to-satellite link. The pointing accuracy will be better than 5° (antenna gain aperture) and shall be increased over the life time with the improvement of the orbit propagator. The final value to be achieved will be 0.5° (tbc). The accuracy of the attitude measurement is determined by

- the magnetometer requirement of 1° knowledge overall
- the need for a ranging link calibration (angular speed of the spacecraft).

After the initial set-up of the constellation flight no regular orbit control is necessary. Only in the case of instable orbit behaviour (too low periselenum) orbit maintenance will become necessary. At the end of the mission, using the achieved knowledge of moon's the gravity field an impact manoeuvre on the moon surface will be executed. The subsatellite attitude and orbit control system makes use of a sensor and actuator concept as depicted in Figure 4.5-36.

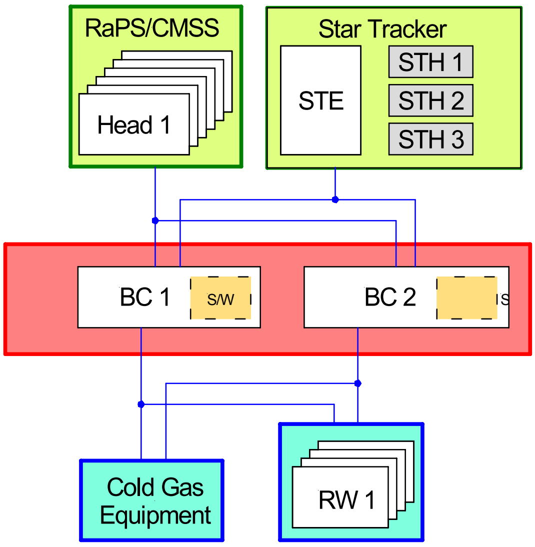


Figure 4.5-36: Subsatellite Attitude and Orbit Control Architecture

For the coarse attitude determination relative to the moon and sun orientation a combination of 6 coarse moon and sun sensors (CMSS) will used that form an integral part of the RaPS instrument. The fine attitude determination takes places making use of three star tracker heads that are operated by one internally redundant electronics package. The orbit determination cannot rely on a dedicated sensor since GPS signals are not available in the moon orbit. The position and relative motion of the spacecraft will be measured using the PRARE-L X-band tracking system whenever the satellite is in the field of view of the PRARE-L ground station. The measurements (in combination with a stepwise refined gravity model of the moon) will be used to continuously update an orbit propagator which serves as the sole on-orbit means for orbit determination.

4.5.3.7 Thermal Control

The main tasks of the subsatellites' thermal control system is the

- Establishment and maintenance of a thermal environment compatible with the requirements of the spacecraft equipment for all mission phases
  - Thermal control (coatings) and thermal insulation (MLI) of the spacecraft to prevent from cooling and overheating, respectively, caused by the external environment (cosmic background, direct solar energy flux, moon reflected solar flux (Albedo), moon emitted infrared energy)
  - Active internal temperature control (electrical heaters, thermistors, etc.)
  - Heat exchange of internal generated waste heat to deep space by space radiators
- Acquisition and transmission of thermal housekeeping data to the on-board data handling system

The overall thermal control principle of the LEO subsatellites is schematically depicted in Figure 4.5-37.

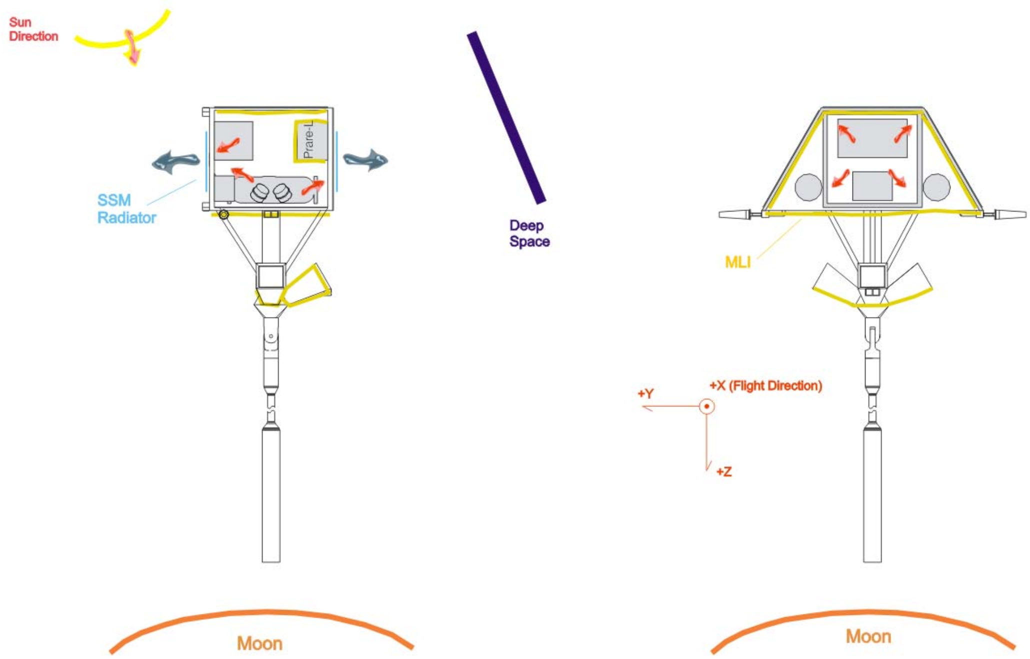


Figure 4.5-37: Subsatellite Overall Thermal Control Concept

The heat generated within the main satellite body will be exchanged among the dissipating units such that an average temperature will prevail inside this volume. An exception is the PRARE-L Ka Front End Equipment which will be partly decoupled from this common thermal environment and will be actively controlled by support heating to a stable temperature. The excess heat will be radiated into space via radiators in flight and anti-flight direction. Due to the high temperature and gradients on the moon surface, the nadir side will not be used for thermal radiation but will be thermally isolated. The star camera heads will be isolated with MLI on all sides.

The magnetometer sensors are installed inside the end section of the boom. The boom is thermally decoupled from the spacecraft body. With a Multi-Layer thermal isolation (MLI) the magnetometers will be kept by passive means within the allowable temperature range and sufficiently stable over time.

The thermal concept of the Ka Front-End (Ka-FE) of the PRARE-L instrument requires a specific thermal control implementation to keep the temperature stability of better than 0.2 K per orbit. The KFE has a dedicated radiator which is trimmed to the appropriate size. The KFE is thermally decoupled to a certain extent from the radiator and from the S/C interior to reduce the temperature variation over the orbit. A thermal buffer mass of approx. 1 kg is implemented to increase the thermal inertia. The remaining temperature variation is reduced to less than 0.2 K per orbit by active heating.

An update of the thermal analysis after the Mid Term Presentation has been performed providing detailed results.

4.5.3.8 Budgets

Mass Budget

The mass budget for the present design of the subsatellite is summarized in Table 4.5-18 applying the rules for mass budget margin as laid down in [AD4]. The harness necessary to interconnect the different items of each instrument are covered in the overall harness figure.

Table 4.5-18: Subsatellite Mass Budget Summary

Item	BEE Mass	Margin	Gross Mass	#	Sum Mass
Structure					**34.8 kg**
Main Structure	21.7 kg	15 %	25.0 kg	1	25.0 kg
Boom	8.5 kg	15 %	9.8 kg	1	9.8 kg
Thermal Control System					**2.9 kg**
Thermal H/W	2.5 kg	15 %	2.9 kg	1	2.9 kg
Electrical Power System					**27.2 kg**
Battery	6.5 kg	5 %	6.8 kg	1	6.8 kg
Roof Solar Panel	2.0 kg	10 %	2.2 kg	1	2.2 kg
Side Solar Panel	2.0 kg	10 %	2.2 kg	2	4.4 kg
Harness	12.0 kg	15 %	13.8 kg	1	13.8 kg
Data Handling System (incuding PCU)					**11.3 kg**
Central Avionics	9.8 kg	15 %	11.3 kg	1	11.3 kg
Attitude & Orbit Control System					**6.0 kg**
Star Tracker Electronics	0.6 kg	5 %	0.6 kg	1	0.6 kg
Star Tracker Head	0.6 kg	5 %	0.6 kg	3	1.8 kg
Reaction Wheels	0.9 kg	5 %	0.9 kg	4	3.6 kg
Propulsion System					**12.0 kg**
Propellant Tank	2.7 kg	5 %	2.8 kg	1	2.8 kg
CG Equipment	8.0 kg	15 %	9.2 kg	1	9.2 kg
PRARE-L					**24.7 kg**
Electronics Unit	12.0 kg	20 %	14.4 kg	1	14.4 kg
Ka-Band Front-End	1.5 kg	20 %	1.8 kg	1	1.8 kg
Ka Antenna	0.3 kg	20 %	0.3 kg	1	0.3 kg
X-Band Front End	5.0 kg	20 %	6.0 kg	1	6.0 kg
X-Band MGA	0.5 kg	20 %	0.6 kg	2	1.2 kg
X-Band LGA	0.5 kg	5 %	0.5 kg	2	1.0 kg
Lunar Mag					**1.4 kg**
Electronics	0.9 kg	10 %	0.9 kg	1	1.0 kg
Sensor	0.2 kg	5 %	0.2 kg	2	0.4 kg
RaPS					**0.6 kg**
Sensor Head	0.1 kg	15 %	0.1 kg	6	0.6 kg

Satellite Dry Mass	**120.9 kg**
Propellant Mass	**7.0 kg**
Satellite Wet Mass	**127.9 kg**

Power Budget

The power budget for the present design of the subsatellite is summarized in Table 4.5-18 applying the rules for power budget margin as laid down in [AD4].

The critical phase obviously will not be the nominal operations but the safe mode when the transmitter of the X-FE is permanently operating and the Ka-FE is off. The additional resulting power will be 11.4 W additionally leading to a complete power demand of 102.2 W. This additional power can be partially compensated by additional switching off of the reaction wheels and the LunarMag instrument. Since the power will not be sufficient during the noon orbits even with the elongated spacecraft body and since the other budgets (power, volume) are increased significantly as well, the purpose of saving budgets that lead to the MTP decision of having the PRARE-L execute TM/TC transition should be abandoned and the originally proposed SpaceTech concept should be strongly reconsidered.

Table 4.5-19: Subsatellite Power Budget Summary (Nominal Operations)

Item	Nominal Power	Duty Cycle	Average Power	Margin	Gross Power	#	Sum Power
Central Avionics	18.0 W	100 %	18.0 W	20 %	21.6 W	1	21.6 W
Reaction Wheels	2.0 W	100 %	2.0 W	5 %	2.1 W	4	8.4 W
Star Tracker	3.0 W	100 %	3.0 W	5 %	3.2 W	1	3.2 W
Heater	50.0 W	15 %	7.5 W	5 %	7.9 W	1	7.9 W
PRARE-L EU	20.0 W	100 %	20.0 W	20 %	24.0 W	1	24.0 W
PRARE-L EU Loss	5.0 W	100 %	5.0 W	20 %	6.0 W	1	6.0 W
PRARE-L Ka-FE	5.5 W	100 %	5.5 W	20 %	6.6 W	1	6.6 W
PRARE-L X-FE TX	25.0 W	40 %	10.0 W	20 %	12.0 W	1	12.0 W
PRARE-L X-FE TX EU Loss	6.3 W	40 %	2.5 W	20 %	3.0 W	1	3.0 W
PRARE-L X-FE RX	1.7 W	100 %	1.7 W	20 %	2.1 W	2	4.2 W
PRARE-L X-FE RX EU Loss	0.4 W	100 %	0.4 W	20 %	0.5 W	2	1.0 W
LunarMag	2.6 W	100 %	2.6 W	10 %	2.9 W	1	2.9 W
				Spacecraft Total:			100.8 W

Data Budget

The data and the accumulation rate for the nominal operation of each LEO subsatellite are summarized in Table 4.5-20. Assuming that per average every second orbit can be used for data transmission to ground and each orbit can be used with 30 % visibility, the required downlink data rate (high data rate link) calculates to 5.8 kbps.

This is slightly higher than the envisaged downlink data rate of 5 kbps. The gap in the data transmission may either be recovered by using additional ground stations from time to time or to store the data until additional visibility from the lunar orbit to ground can be established (orbit normal vector pointing towards Earth).

Table 4.5-20: Subsatellite Data Rate Budget

Item	No Contact	Contact
Satellite Telemetry Data	100 bps	100 bps
Magnetometer Science Data	212 bps	212 bps
PRARE-L ISL Data	400 bps	400 bps
PRARE-L Tracking Data	-	400 bps
Intermediate Sum	712 bps	1112 bps
Formatting Overhead (12 %)	85 bps	122 bps
Spacecraft Total:	797 bps	1245 bps

Link Budget

The link budget for the subsatellite links from Earth to spacecraft (uplink) and spacecraft to Earth is summarized in Table 4.5-21 and Table 4.5-22, respectively. The uplink is assumed at 7.21 GHz, a bandwidth of 15 MHz and a data rate of 4 kbps.

Table 4.5-21: Subsatellite Uplink Budget

	LGA	MGA
Power Output	13,16 dBW	13,16 dBW
Antenna Gain	63 dBi	63 dBi
EIRP	76,16 dB	76,16 dB
Power Flux Density	-133.26 dBW/4kHz/m^2	-133.26 dBW/4kHz/m^2
C/N0	49.62 dBHz	54.12 dBHz
Required C/N0	36.00 dBHz	36.00 dBHz
Locking / Ranging Margin	13.62 dB	18.12 dB
Eb/N0	12.6 dBHz	17.1 dBHz
Required Eb/N0	9.6 dBHz	9.6 dBHz
Data Margin	3.00 dB	7.50 dB

The down-link is assumed at a transmit frequency of 8.48 GHz, a band-width of 15 MHz and a data rate of 5 kbps.

Table 4.5-22: Subsatellite Data Rate Budget

	LGA	MGA
Power Output	7 dBW	7 dBW
Antenna Gain	-3.0 dB	+1.5 dB
EIRP	3.38 dBW	7.88 dBW
Power Flux Density	-206.03 dBW/4kHz/m^2	201.53 dBW/4kHz/m^2
C/N0	46.29 dBHz	50.79 dBHz
Required C/N0	36.00 dBHz	36 dBHz
Locking / Ranging Margin	10.29 dB	14.79 dB
Eb/N0	8.29 dB/Hz	12.8 dB/Hz
Required Eb/N0	0.3 dB/Hz	0.3 dB/Hz
Data Margin	7.99 dB	12.50 dB

4.6 Ground Segment

4.6.1 Ground Segment Overview

4.6.1.1 Ground Segment Operation Concept

The LEO Operations Concept is based on a Mission Operation System (MOS) as illustrated in Figure 4.6-1

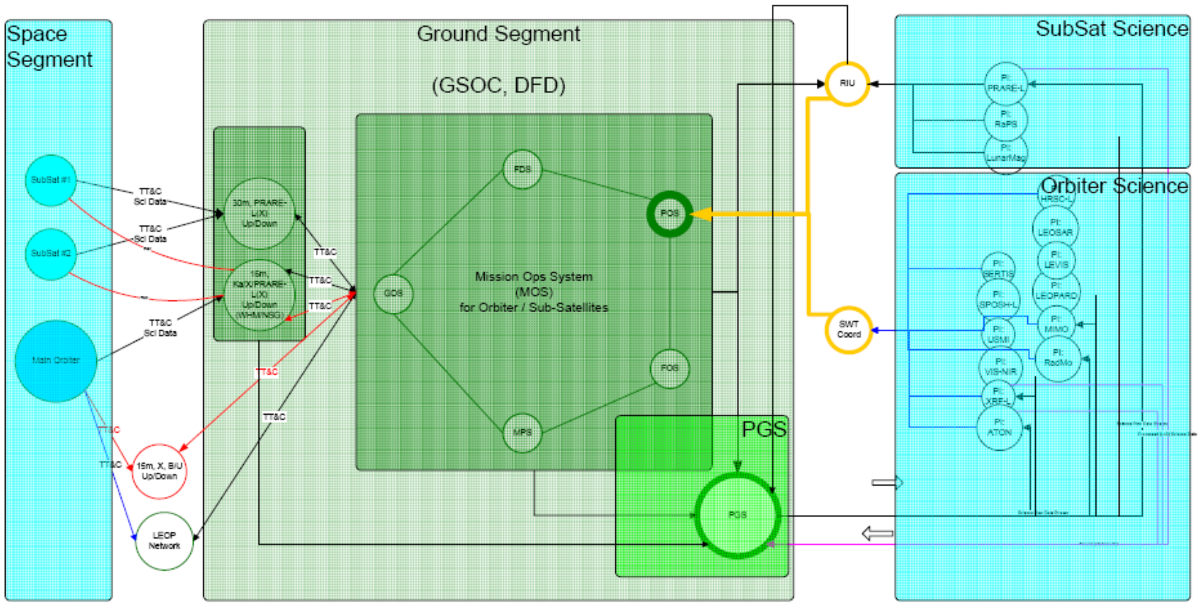


Figure 4.6-1 LEO Operations Concept

The MOS is based on the

- Flight Operations System (FOS)
- Payload Operations System (POS)
- Mission Planning System (MPS)
- Flight Dynamics System (FDS)
- Ground Data System (GDS)

It interfaces to the Space segment (the main satellite plus two sub-satellites), the Payload Ground System (at DFD) and the instrumenters, coordinated by the SWT and RIU (for the instruments on the sub-satellites), respectively.

As the overall control of the satellite platforms is within the ground segment it was identified that MOS should be able to have all commanding and monitoring under control, i.e. to condense and shorten data flow within the MOS to improve security and increase reaction times for bus and payload operations. While the spacecraft bus is a server to the payload, the payload and bus systems are not functioning independently. Concurrency among the different payloads, on the one hand, bus system status constraints, on the other hand, may interact. Therefore, central task of the MOS is to always optimize the mission schedule in terms of the overall system status including the ground segment status itself.

Figure 4.6-1 also indicated the data flow from the 3 satellites to the two foreseen antennas (16m Ka/X-Band and 30m X-Band), to MOS and further to RIU/SWT and PGS for archiving.
Additional antennas are foreseen for backup and LEOP.

The links between the satellites and Ground Stations is indicated in more detail in Figure 4.6-2.

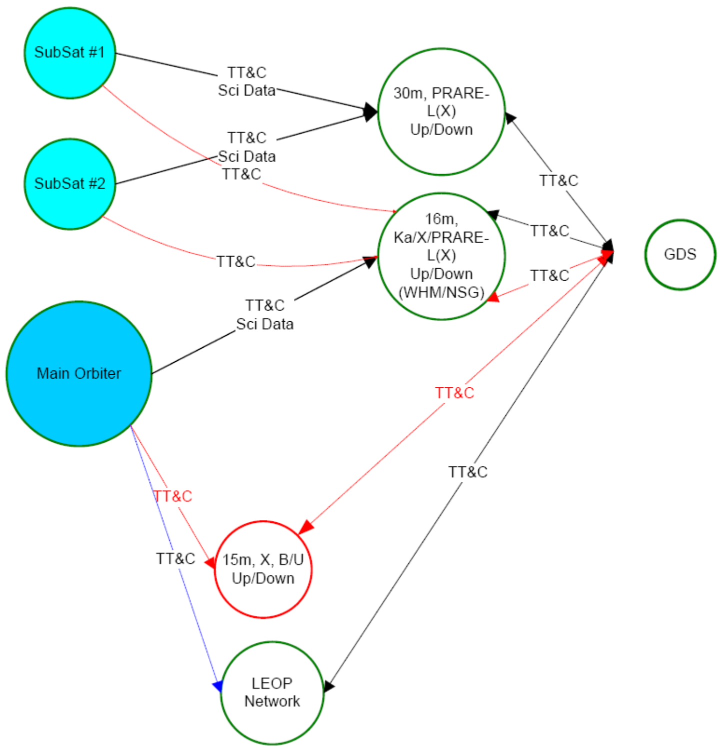


Figure 4.6-2 TT&C and SciData links between LEO satellites and ground stations.

It needs to be stressed, that it is planned to use as much as possible existing structures and eventually integrate LEO operations into the currently practiced multi-mission-operations concept.

4.6.1.2 Ground Segment Management Concept

The overall responsibility for the LEO Ground Segment is at GSOC, with certain elements at the DFD and at RIU.

The Ground segment of PRARE-L, including the H/W as implemented into the ground station is under responsibility of RIU. Here also the science coordination for the instruments aboard the sub-satellites takes place.

The corresponding science coordination for the experiments on the main satellite is taken care of by the LEO-SWT.

Due to the complex interface between SWT and RIU with the MOS, the POS (Payload Operations System) has been introduced into the MOS structure (Figure 4.6-1).

Data archiving is under the responsibility of the PGS at DFD.

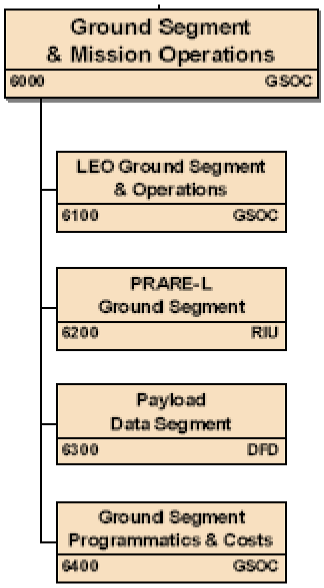


Figure 4.6-3 WBS indicating ground segment responsibilities

Figure 4.6-3 shows the excerpt of the WBS regarding the Ground segment, indicating WP´s with the respective responsibilities at GSOC, DFD and RIU.Ground Stations

4.6.1.3 LEOP and Transfer Ground Stations

The LEOP Ground Station network is used to have enough contact with the satellite during the LEOP and Transfer Phase to be able to monitor and control the spacecraft to bring in a secure way to its end orbit.

The LEO LEOP has to be made in X-Band, so the choice of the LEOP ground stations is more reduced as in the case of a classic S-Band LEOP. The transfer phase will be realised by a similar ground station network and is assimilated as part of the LEOP for the purpose of that document.

Here is our proposition for a basis LEO LEOP ground station network:

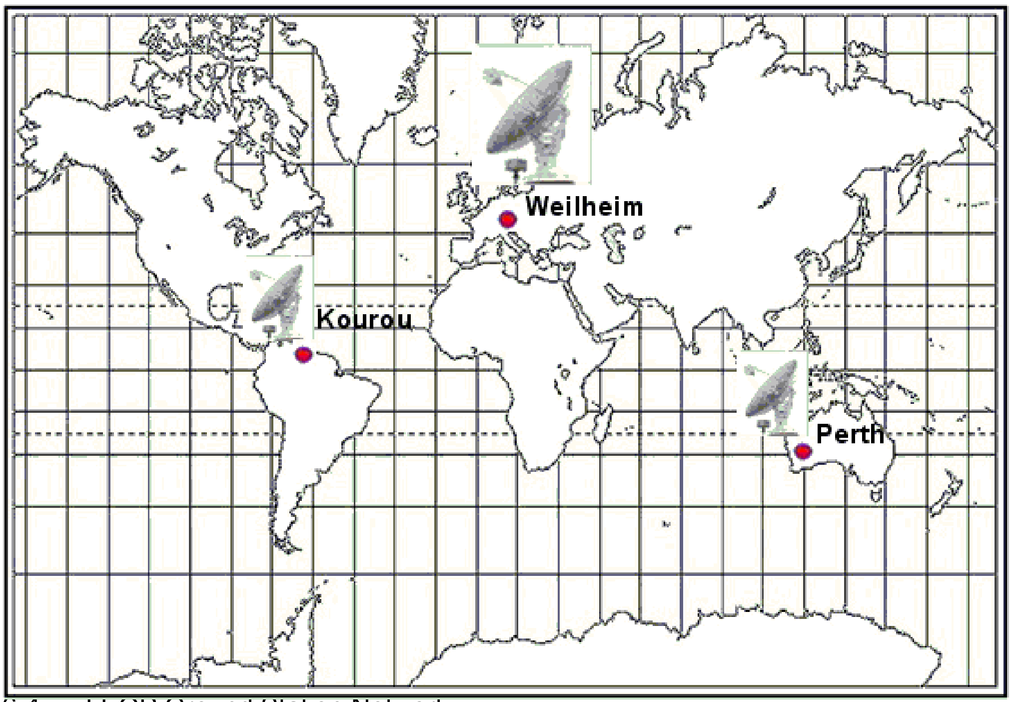


Figure 4.6-4 LEOP Ground Station Network

The choice of these 3 ground stations is based on a global coverage in the longitude of Europa, Amerika and Asia.

These 3 ground stations will not be enough and following X-Band stations could be added if needed:

- Kourou (French Guyana) – ESA
- Kagoshima (Japan) – JAXA DSN
- Goldstone (USA) – NASA DSN
- Canberra (East Australia) – NASA DSN

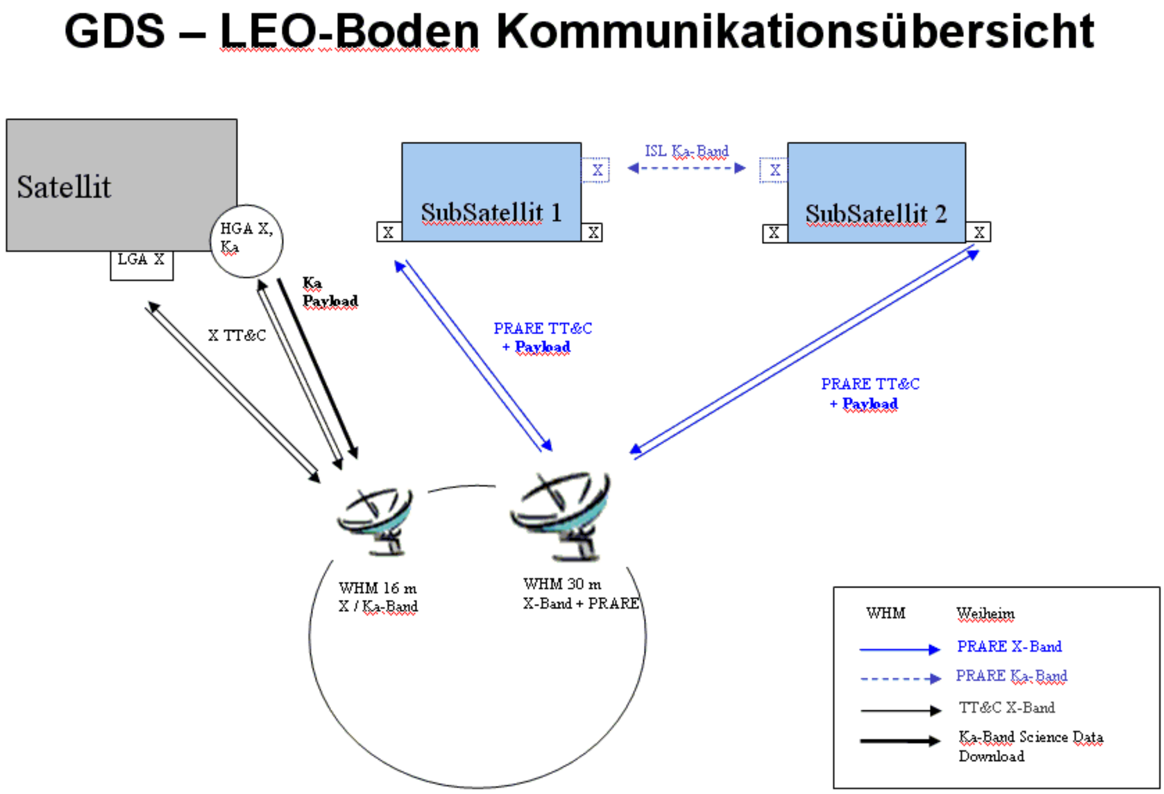
For critical phases, the contact to the satellite has to be realised with 2 ground stations pointing simultaneously at the spacecraft. The ground station list provided before can be used to assure that this requirement can be complied.

Most of the ground station provided in the list are DSN (Deep Space Network) antenna. These Antenna are relative slow and it has to be analysed in detail in a later phase which antenna can be used for which part of the LEOP. For Transfer, there should be no problem because the satellite is not moving so rapidly over the antenna as by the LEOP.

Other factors playing an important role by LEOP network are the costs. It is better to take bundles of ground stations (several ground stations belonging to the same agency) as a lot of different agencies antenna. All these facts explain why it is too early in Phase A to make more than a ‘very draft’ LEOP network. In Phase B a detailed analyse of which ground station can be used in which period of LEOP and Transfer has to be done (from the orbit point of view but also from the costs one).

4.6.1.4 Main Satellite Ground Station

Here is a general overview of the LEO Satellite(s) – Ground Station communication:



TT&C communication

The TT&C communication with the main satellite is done with X-Band for uplink and downlink.

The DLR ground station Weilheim is the main station in the baseline and a new 16m X-/Ka-Band Antenna is planned to be used (an option is to build the new Antenna in the DLR ground station Neustrelitz). The frequency used is a deep space frequency (deep space type A) for X-Band. The Weilheim ground station has already a corresponding licence.

Ranging will also be done via X-Band in parallel to TM monitoring.

Here are the Characteristics of this new antenna:

- X-Band (uplink & downlink) for TT&C communication with main satellite
- Ka-Band (downlink) for Payload communication with main satellite
- Antenna gain: It is difficult to define a real antenna diameter and of course a real antenna gain because only the manufacturer can definitively do that. But to enable the project to make a link

budget calculation, we have made some assumption on a required Gain for the new antenna.

- The antenna needs a strong data connection to PGS (Payload Ground System). An option will be to build the Ka-Band Antenna directly in Neustrelitz.
- The PRARE-L System should be installed in the antenna because of using this antenna as a backup for the sub-satellites

Payload communication

The Payload communication is done with Ka-Band (downlink)

The DLR ground station Weilheim is the main station in the baseline and a new 16m X-/Ka-Band Antenna is planned to be used (an option is to build the new Antenna in the DLR ground station Neustrelitz).

Interfaces with Control Center

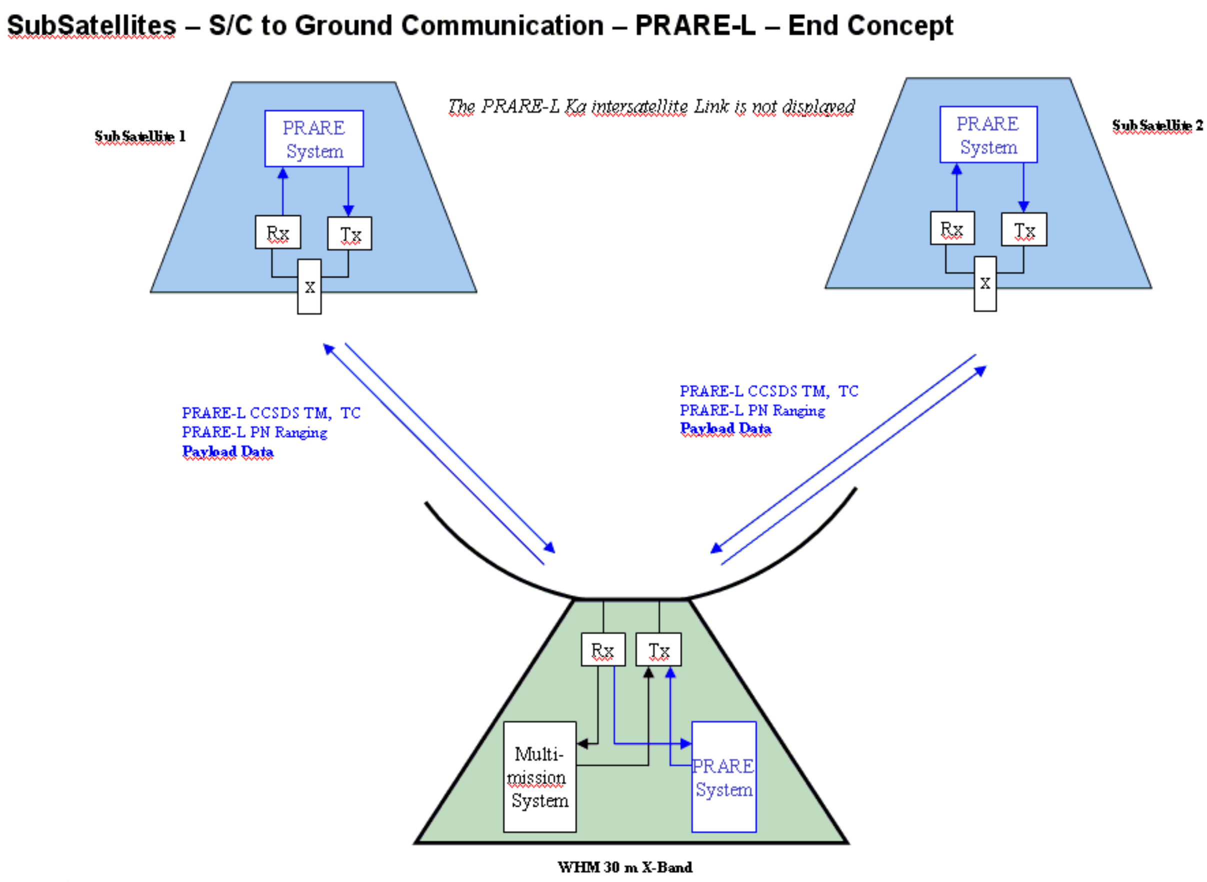
Following interfaces were identified between the ground station Weilheim (or Neustrelitz as option) and the GSOC Control Center for the purpose of the main satellite (16m Ka-/X-Band antenna).

- TMTC Realtime Interface
- Offline Data Interface on FTP basis (Pointing informations, Angle Data measurements, Orbit predictions, tracking data, a.s.o.)
- Payload Data Interface to PGS

4.6.1.5 Sub-Satellite Ground Station

For a general overview of the LEO Satellite – Ground Station communication, please refer to chapter 4.6.1.4:

Here is a detailed concept about how the sub-satellites to ground communication could be made. It should use the PRARE-L system as main communication system. The reason for this decision is that this system allows a very precise ranging needed to process the gravimetry measurements of the sub-satellites. A secondary "classic" communication system based on CCSDS standards for LEOP and Emergency cases was proposed from GSOC but has not be retained for the baseline. This decision correspond for us to a bigger risks of loosing the 2 Sub- satellites if some problem occurs with the PRARE-L communication system that is only a "prototype". In case of an emergency, it is not possible to add contact from other ground stations in the world because they don't have a PRARE-L system.



TT&C communication

The TT&C communication is done with X-Band for uplink and downlink. The TT&C system will be a special PRARE-L Ranging & TMTC system developed for that project and supporting the CCSDS standards for frame / packet definition.

The special PRARE-L Ranging & TMTC system is needed for a very precise ranging of both sub-satellites for fine processing of the gravitation payload information measured on board.

The ground station form DLR Weilheim is the main station in the baseline and the 30m Deep space X-Band Antenna S68 is planned to be used. The frequency used is a deep space frequency for X-Band. The Weilheim ground station has already a corresponding licence.

Ranging will also be done via X-Band in parallel to TM monitoring using PRARE-L system.

Here are the Characteristics required for the 30m antenna in Weilheim:

- The antenna should communicate simultaneously (at the minimum for PRARE ranging) with both sub-satellites.
- The PRARE-L System should be installed in the antenna feed.

Payload communication

The Payload communication is done with X-Band (downlink) using the same system as for TT&C communication

The DLR ground station Weilheim is the main station in the baseline and the 30m Deep space X-Band Antenna S68 is planned to be used.

Interfaces with Control Center

Following interfaces were identified between the ground station Weilheim and the GSOC Control Center for the purpose of the sub-satellites (30m X-Band antenna).

- TMTC Realtime Interface (through the PRARE-L TX/RX box)
- Offline Data Interface on FTP basis (Pointing information, Angle Data measurements, Orbit predictions, tracking data, a.s.o.) + Similar FTP Interface for processed tracking Data from RIU to GSOC
- Payload Data Interface on a FTP basis

4.6.2 Flight Dynamics

GSOC Flight Dynamics will conduct all flight dynamics operations according to the predefined Sequence of Events from separation from the launcher, transfer to the Moon, injection into the target orbit and keeping of the operational orbit around the Moon. These tasks are described in the following sections.

4.6.2.1 Attitude Determination and Analysis

Three-axis attitude determination and analysis will be performed throughout all mission phases and for all satellites (i.e. main and sub-satellite) based on dumped AOCS housekeeping data comprising of attitude sensor measurements.

During Commissioning and Operational Phases the precise attitude of all satellites will be determined from star-tracker measurements in order to generate an Attitude Product for Science Community with specified accuracy and latency. Aiming on more accurate results, optimal quaternion combination will be applied whenever simultaneous measurements from multiple star-trackers are available.

4.6.2.2 Orbit Determination and Maneuver DV Estimation

Prior to the launch, an initial acquisition analysis will be performed based on the injection dispersion (as specified by launch agency) and the involved ground stations to identify possible problems acquiring the satellite during the first orbits.

During all phases of the mission, 2-way range and 2-way Doppler data received from all involved ground stations are pre-processed and then used for orbit determination.

Especially during the first phase of the mission after separation from the launcher it is important that sufficient tracking data are available as soon as possible. Therefore at least two ground stations should have visibility to the spacecraft in this critical phase of first acquisition.

It is important to perform this first orbit determination as fast as possible, because otherwise the launcher’s injection dispersion could cause problems to recover the satellite during following contacts. On the other hand, an exact orbit determination is essential for the planning of orbit maneuvers to achieve the target transfer trajectory to the Moon.

In case of the routine sub-satellites operation range and Doppler tracking data will be supplemented by PRARE-L data to improve the operational orbit determination accuracy. Based on the comparable Lunar Prospector mission position accuracies of < 20 m in radial, < 400 m in along-track and < 300 m in normal direction (RMS) can be expected when applying the LP100J gravity model.

The velocity increment (DV) and thrust vector direction of executed orbit maneuvers will be estimated along with the orbit determination as input for the calibration of the thrusters which is important for the planning of following orbit maneuvers.

GSOC Flight Dynamics will provide posteriori orbit determination as well as orbit prediction products to the scientific community and mission planning system resulting from operational orbit determination within a latency and accuracy to be specified. High-precision orbit products will be re-processed based on gravity field model upgrades as available by PRARE-L experiment results or other Lunar gravity field missions.

4.6.2.3 Operations Support

Based on the orbit determination results, the following orbit related products will regularly be generated:

- Pointing data for Weilheim ground station
- Orbit data for other ground stations
- Events for MOS and GDS
- Orbit related information
- On-board orbit propagator updates for S/C upload

4.6.2.4 Orbit Control

In general, based on the orbit determination results, the orbit parameters of "LEO" main and sub-satellites are monitored and maneuvers are planned in order to fulfil the requirements on the control accuracy with respect to pre-defined reference orbits.

The executed maneuvers will be calibrated within the orbit determination process in order to assess the propulsion system performance, which again is required as an input to the maneuver planning process (i.e. command generation), and determine the used propellant mass. The latter will allow for the computation of actual S/C mass and Center of Gravity (based on algorithms provided by S/C manufacturer).

Main Satellite

The cruise phase starts with S/C separation from launcher (about 90 minutes after launch) on a trans-lunar trajectory. Within the early cruise phase (about 6 hours duration) the S/C performs sun and Earth acquisition. Here initial planning of trajectory correction maneuvers is performed.

The first trajectory correction maneuver (TCM1) is planned 6 hours after trans-lunar injection (TLI) performed by Fregat stage. This TCM1 compensates the dispersions caused by the Soyuz / Fregat performance. Therefore an exact first orbit determination and a TCM1 maneuver planning is to be performed immediately after separation. For this purpose at least 2 ground stations shall be available ensuring range and Doppler tracking of the spacecraft.

The performance of TCM1 is estimated by orbit determination and a second maneuver (TCM2) is planned ca. 24 hours after TCM1 in order to compensate execution errors of this first maneuver.

A third correction maneuver is planned ca. 24 hours prior to Lunar Orbit Insertion (LOI) based on exact orbit determination during the cruise phase (ca. 3-4 days) after TCM2 and calibration of the second maneuver.

After execution and calibration of TCM3 the Lunar Orbit Insertion maneuvers (LOI1 to LOI3) are planned targeting on a circular, 100 km altitude, 85 deg inclination orbit. The executed LOI maneuvers are calibrated by orbit determination and a fine tuning of the following maneuver(s) is done, as applicable.

The achieved operational orbit will be kept throughout the Commissioning Phase (about 60 days duration) with tolerable variations in altitude (i.e. 100±25 km) and inclination (i.e. 85±0.5 deg). Note that within this phase the sub-satellites are deployed.

Sub-satellites

The sub-satellites are deployed from the main satellite during its commissioning phase (i.e. 100 km altitude and 85 deg inclination).

Within a sub-sequent 20 days commissioning the sub-satellites will be maneuvered to acquire their target orbit, which is currently specified to have a 50±30 km altitude and 85 deg inclination. Furthermore, the sub-satellites will be separated in along-track direction by about 60 to 120 km, where the distance shall be kept with 5 to 10 km control accuracy over maneuver-free periods of at least 28 days. The nominal mission duration is 3 years with optional extended mission within same orbit.

Following each 28+ days drift phase a brief orbit maintenance phase should be foreseen in order to control the absolute and relative motion. In case the absolute orbits (i.e. altitude, inclination) are not subject to change, the orbit maintenance could be performed with a single sub-satellite relative to the other (i.e. adjusting mean along-track distance and relative eccentricity vector).

Currently the following control aspects are not known:

- How is the deployment done? What are the changes in orbital velocity of sub-satellites implied by the separation process?
- Is the mean along-track distance an input (e.g. from PRARE-L PI or mission planning) or can it be independently adjusted by FDS within the required control band (i.e. 60 to 120 km)?
- The mission analysis report states that attitude control thruster firings are foreseen and perturb the orbit such that cm/s or mm/s control maneuvers are required to counter-balance their effect on the orbit. On the other side the PRARE-L experiment requires maneuver-free data acquisition periods of 28 days. It has to be clarified whether these correction maneuvers and eventually the attitude pulses itself significantly perturb the measurement campaign.

It further has to be noted that on-board AOCS nominal and safe mode concepts featuring thruster firings for purpose of attitude control are considered as unfavourable because uncontrolled change of absolute and relative (i.e. between sub-satellites) orbits will perturb the orbit prediction accuracy and further might result in increased risk of collision.

The sub-satellites mission ends with their impact on lunar surface. Final maneuvers might be planned in order to control impact location and time.

4.6.3 PRARE-L Ground Segment

The primary objective of the PRARE-L experiment is the determination of the lunar gravity field at high accuracy and high harmonic resolution using satellite-to-satellite and satellite-to-ground (Earth) tracking data.

The basic measurements are the determination of the relative velocity between the two sub-satellites. Satellite positioning and orbit determination of the subsatellites is achieved by range and range-rate measurements between the two satellites and a tracking station on Earth. The subsatellites need to be visible from the tracking station.

The PRARE-L experiment consists of a space segment and a ground segment.

The space segment consists of two identical instruments on both subsatellites and realizes an interspacecraft link at Ka-band and a space-ground link at X-band for each subsatellite. This requires dedicated equipment in a tracking station (ground segment) which is considered as an integral part of the PRARE-L instrument.

The task of the PRARE-L ground segment is the tracking of both subsatellites from ground, to record range and range rate at X-band, to upload telecommands for the subsatellites and its payload and ephemeris information for the PRARE-L space segment, to receive telemetry from the instruments and the sub-satellite house-keeping, to receive and to transpond time stamps from the PRARE-L space segment on the sub-satellites and the ground timing system, respectively.

The dedicated ground station antenna is the 30-m dish in Weilheim-Lichtenau which needs to be equipped with the PRARE-L ground segment elements. The PRARE-L inter-spacecraft link between both subsatellites shall operate continuously. The spacecraft-to-ground radio link can only be operated when the subsatellites are visible from the ground station site. Three to five contact or operation periods per day may be feasible assuming an subsatellite orbital period of two hours and a visibility of the Moon at Weilheim of six to ten hours.

A Abbreviations

3D	three dimenional
AC	Alternating Current
ACS	Avionics Control System
AD	Applicable Document
ADC	Analog-to-Digital Converter
ADD	Architectural Design Document
ADEV	Allan DEViation
ADP	Acceptance Data Package
AEM	Apogee Raising Module
AI	Action Item
AIT	Assembly, Integration and Test
AIV	Assembly, Integration and Verification
AM/PM	Amplitude Modulation/Phase Modulation
AMF	Apogee Motor Firing
amu	Atomic Mass Unit
AOC	Attitude & Orbit Control
AOCE	Attitude and Orbit Control Electronics
AOCS	Attitude and Orbit Control System
APD	Avalanche photodiode
AR	Acceptance Review
ARC	Austrian Research Center
ARC	Atlantic Research Corporation
ASG	Astrium GmbH
ASIC	Application Specific Integrated Circuit
AST	Astos Solutions GmbH
ASTOS	AeroSpace Trajectory Optimization Software
ASW	Address and Synchronisation Word
ATON	Autonomous Terrain-based Optical Navigation
ATS	Acceptance Test Specification
BAPTA	Bearing and Power Transfer Assembly
BAQ	Block Adaptive Quantizer
BAT	Battery
BB	Black Body
BBM	Broadband Model

A Abbreviations

BC	Board Computer
BDR	Baseline Design Review
BoL	Begin of Life
BPS	Bits per Second
C/O	Check-Out
CA	Central Avionics
CADU	Channel Access Data Unit
CCD	Charge coupled device
CCN	Contract Change Notice
CCSDS	Consultative Committee for Space Data Systems
CDC	Concurrent Design Centre
CDR	Critical Design Review
CE	Concurrent Engineering
CESS	Coarse Earth and Sun Sensor
CFI	Customer Furnished Item
CFK	Carbon Fibre Reinforced Plastic
CGN	ColoGNe
CHAMP	CHAllenging Mini-satellite Payload
CI	Configuration Item
CIL	Critical Items List
CIP	Catalogue Interoperability Protocol
CM	Configuration Management
CMOS	Complementary Metal Oxide Semiconductor
CMSS	Coarse Moon & Sun Sensor
CN	Change Notice
COC	Certificate of Conformance
COTS	Commercial Off The Shelf
CP	Chemical Propulsion
CPA	Central Parts Procurement Agency
CPPS	Chemical Propulsion & Power System
CPU	Central Processing Unit
CR	Change Request
CRP	Carbon Reinforced Plastics
CS	Coax Switch
CSG	Centre Spatial Guyanais
CSM	Communication System Monitoring

D	Diplexer
D&DRL	Data and Documentation Requirements List
D/L	Downlink
DAC	Digital-to-Analog Converter
DC	Direct Current
DCL	Declared Component List
DD&V	Design, Development and Validation Plan
DDF	Design Definition File
DDP	Design & Development Plan
DEC	Decoder
DET	Direct Energy Transfer
DFD	Deutsches Fernerkundungsdatenzentrum
DFS	Deutscher Fernmelde Satellit Kopernikus
DHS	Data Handling System
DIL	Deliverable Items List
DIMS	Data Information and Management System
DLR	Deutsches Zentrum für Luft- und Raumfahrt
DLR-RY	DLR Institute of Space Systems
DML	Declared Materials List
DN	Discrepancy Notice
DNEL	Disable Non-Essential Loads
DoD	Depth of Discharge
DoF	Degree of Freedom
DPL	Declared Process List
DQA	Development, Qualification and Acceptance
DRL	Document Requirement List
DSN	Deep Space Network (NASA)
DSPG	Distributed Single Point Grounding
E-Box	Electronics Box
ECI	Earth Centered Inertial
ECSS	European Cooperation for Space Standardization
EDAC	Error Detection and Correction
EDC	Effective Date of Contract
EGSE	Electrical Ground Support Equipment
EIRP	Equivalent Isotropic Radiated Power
EM	Engineering Model

A Abbreviations

EMC	ElectroMagnetic Compatibility
EMI	Electro Magnetic Interference
ENC	Encoder
EOC	End Of Contract
EoL	End of Life
EP	Electric Propulsion
EPPS	Electric Propulsion & Power System
EPS	Electric Power System
EPU	Electrical Power Unit
EQSR	Equipment Qualification Status Review
ESA	European Space Agency
ESD	Electrostatic Discharge
ESTEC	European Space Research and Technology Centre
EU	Electronics Unit
FDIR	Failure Detection Isolation and Recovery
FDS	Flight Dynamics System
FE	Front End
FEEP	Field Emission Electric Propulsion
FiFo	First in First out
FM	Flight Model
FMCR	Flight Model Completion Review
FMECA	Failure Mode, Effects and Criticality Analysis
FOS	Flight Operations System
FOV	Field of view
FPA	Focal Plane Assembly
FPGA	Field Programmable Gate Array
FRR	Flight Readiness Review
FTM	Factory Test Meeting
FTP	File Transfer Protocol
G/T	Gain over Temperature
GCR	Galactic Cosmic Ray
GCS	Ground Control System
GDS	Ground Data System
GDU	Gas-Distributing Unit
GEO	GEOstationary orbit
GFZ	Helmholtz Centre Potsdam, GFZ German Research Centre for Geosciences

GIE	Gridded Ion Engine
GNC	Guidance, Navigation and Control
GOCE	Gravity and steady-state Ocean Circulation Explorer
GRACE	Gravity Recovery and Climate Experiment
GS	Ground Segment
GSD	Ground Sampling Distance
GSE	Ground Support Equipment
GSN	Ground Station Network
GSOC	German Space Operations Center
GTO	Geostationary Transfer Orbit
GUI	Graphic User Interface
HBK	Handbook
HEMP(T)	High Efficient Multistage Plasma (Thruster)
HET	Hall Effect Thruster
HET	High Energy Telescope
HGA	High Gain Antenna
HK	HouseKeeping
HLSST	High-Low Satellite-to-Satellite Tracking
HMI	Human Machine Interface
HPA	High Power Amplifier
HPC	High Priority Command
HRSC-L	High Resolution Stereo Camera - Lunar version
HRWS	High Resolution Wide Swath
HTML	HyperText Mark-up Language
HTR	Heater
HTTP	HyperText Transfer Protocol
HVC	High Voltage Converter
HW	Hardware
I/F	Interface
I/O	Input / Output
I/Q	In-phase / Quadrature-phase
IC	Integrated Circuit
ICD	Interface Control Data/Document
ICU	Instrument Controller Unit
IDA	Institute of Computer and Communication Network Engineering of TU Braunschweig
IF	Interface

A Abbreviations

IFOV	Instantaneous FOV
IMU	Inertial Measurement Unit
INV	Inventory
IOAR	In Orbit Acceptance Review
IOT	In Orbit Test
IP Core	Intellectual Property Core
IPFD	Input Power Flux Density
IR	Infrared
ISL	Inter-Satellite Link
ISP	Instrument Source Packet
ITT	Invitation to Tender
ITU	International Telecommunication Union
JOP	Jena-Optronik GmbH
Ka-FE	Ka-Band Front End
KIP	Key-Inspection Point
KO	Kick Off
L0	Level-0 Product
L1	Level-1 Product
L2	Level-2 Product
LAN	Local Area Network
LC3	Linear Charge-Current Control
LCL	Latching Current Limiter
LCL	Latching Current Limiter
LEO	Lunar Exploration Orbiter
LEO	Low Earth Orbit
LEOP	Launch and Early Orbit Phase
LEOPARD	Lunar Exploration Orbiter Dust Particle Detector
LEOSAR	Lunar Exploration Orbiter Synthetic Aperture Radar
LET	Linear Energy Transfer
LEVIS	Lunar Exploration Video Imager System
LGA	Low-Gain Antenna
LLO	Low Lunar Orbit
LLSST	Low-Low Satellite-to-Satellite Tracking
LNA	Low Noise Amplifier
LOI	Lunar Orbit Insertion
LRR	Launch Readiness Review

LRT	Lehrstuhl für Raumfahrttechnik, TU München
LSB	Least Significant Bit
LTI	Lunar Transfer orbit Insertion
LTO	Lunar Transfer Orbit
LunarMag	Magnetometer
LVC	Low Voltage Converter
LZ77	Lempel Ziv 77
MAG	MAGnetometer
MAG-E	Magnetometer Electronics
MAG-S	Magnetometer Sensor
MCP	Micro Channel Plate
MCT	Mercury Cadmium Telluride
MDR	Mission Definition Review
MEM	Memory
MFB	Multi-Functional Boards
MGA	Medium Gain Antenna
mGal	10-5 m/s2
MIGIM	Micro-Gradiometer Instrument for Moon
MIMO	Microwave Instrument for a Moon Orbiter
MIP	Mandatory Inspection Point
MJD	Modified Julian Date
MLI	Multi-Layer Insulation
MMB	Mass Memory Board
MoM	Minutes of Meeting
MORO	Moon ORbiting Observatory
MOS	Mission Operations System
MPB	Main Processor Board
MPD	Magneto Plasma Dynamic Thruster
MPS	Mission Planning System
MSB	Most Significant Bit
MTF	Modulation Transfer Function
n.a.	not applicable
NAND	Not AND
NASA	National Aeronautics and Space Administration
NCR	Non Conformance Report
NEDT	Noise Equivalent Difference Temperature

A Abbreviations

NESZ	Noise Equivalent Sigma Zero
NETD	Noise Equivalent Temperature Difference
NiCD	Nickel Cadmium
NiH2	Nickel Hydrogen
NIR	Near Infrared
NRT	Near-RealTime
NSG	NeuStrelitz Ground station
NSSK	North/South Station Keeping
NTNS	Non-Thermal Neutron Sensor
NTP	Network Time Protocol
OAIS	Open Archival Information System
OBC	On-Board Computer
OBDH	On-Board Data Handling
OBT	On-Board Time(r)
OP	OberPfaffenhofen
OPD	Optical Path Difference
OTF	Optical Transfer Function
OTS	Off The Shelf
P/F	Platform
P/L	Payload
PA	Product Assurance
PCB	Printed Circuit Board
PCD	Power Control and Distribution module
PCDU	Power Conditioning and Distribution Unit
PCM	Pulse Code Modulation
PcR	Processing Request
PCU	Power Conditioning/Control Unit
PDF	Portable Document Format
PDR	Preliminary Design Review
PdR	Production Request
PDS	Payload Data Segment / Planetary Data System / Public Data System
PDT	Payload Data Transmission
PDU	Power Distribution Unit
PFM	Proto Flight Model
PGS	Payload Ground System
PI	Principal Investigator

PLL	Phase Lock Loop
PM	Progress Meeting
PMC	Payload Management Computer
PMM	Payload Mass Memory
PMP	Project Management Plan
PMT	Photo-Multiplier Tube
PN-code	Pseudo Noise Code
POS	Payload Operations System
PPDU	Payload Power Distribution Unit
PPS	Pulse Per Second / ProPulsion Subsystem
PPT	Peak Power Point Tracking
PPT	Pulsed Plasma Thruster
PPU	Power Processor Unit
PR	Power Relay / Pressure Regulator
PRARE-L	Precise Range And Range-rate Equipment – Lunar version
PRARE-L GS	PRARE-L Ground Segment at Weilheim
PRARE-L OC	PRARE-L Operation Center at RIU-PF
PRF	Pulse Repetition Frequency
PROM	Programmable Read Only Memory
PRR	Preliminary Requirement Review
PSF	Point Spread Function
PSK	Phase Shift Keying
PSM	Processing System Management
PSS	Payload Support System
PSU	Power Supply Unit
QA	Quality Assurance
QC	Quality Control
QM	Qualification Model
QPSK	Quaternary Phase Shift Keying
R&D	Research and Development
RAAN	Right Ascention of the Ascending Node
RadMo	Radiation Monitor
RAM	Random Access Memory
RaPS	Radiation Pressure Sensor
RD	Reference Document
RF	Radio Frequeney

A Abbreviations

RFD	Request For Deviation
RFFE	Radio Frequency Front End
RFI	Request For Information
RFP	Request For Proposal
RFQ	Request For Quotation
RID	Review Item Discrepancy
RIT	Radiofrequency Ion Thruster
RIU	Rheinisches Institut für Umweltforschung
RIU-PF	Rheinisches Institut für Umweltforschung – Abt. PlanetenForschung
ROI	Region of Interest
ROIC	Read Out IC
ROM	Rough Order of Magnitude
RS	Reed Solomon
RSS	Root Sum Squared
RW	Reaction Wheel
RX	Receiver
Rx	Receive
S/A	Solar Array
S/C	Spacecraft
S/S	SubSystem
S/W	SoftWare
S3R	Sequential Switching Shunt Regulator
SA	Solar Array
SADA	Solar Array Drive Assembly
SADM	Solar Array Drive Mechanism
SAR	Synthetic Aperture Radar
SBF	Self Blocked Filter
SC	Splitter / Combiner
SCC	Satellite Control Centre
SCM	Simulator Completion Review
SDRAM	Synchronous Dynamic Random Access Memory
SEE	Single Event Effect
SEL	Single Event Latch up
SELENE	SELenological and ENgineering Explorer
SEP	Solar Energetic Particle
SERTIS	SElenological Radiometer and Thermal Infrared Spectrometer

SET	Single Event Transient
SEU	Single Event Upset
SGEO	Small Geostationary Platform
SGG	Satellite Gravity Gradiometry
SHF	Super High Frequency
SIT	Satellite Integration Test
SM	Structural Model
SMPS	Switch Mode Power Supply
SNR	Signal to Noise Ratio
SoC	System on a Chip
SOC	Switch-Over Control
SOI	Sphere of Influence
SPA	Software Product Assurance
SPE	Solar Particle Event
SPL	Single Point Failure List
SPOSH	Smart Panoramic Optical Sensor Head
SPT	Satellite Performance Test / Stationary Plasma Thruster
SPU	Satellite Processing Unit
SpW	SpaceWire
SQA	Software Quality Assurance
SRR	System Requirement Review
SSD	Spatial/Spectral Sampling Distance
SSD	Solid-State Detector
SST	Satellite-to-Satellite Tracking
STC	Star Tracker Camera
STE	Star Tracker Electronics
STH	Star Tracker Head
STS	Space Transportation System
STS	Short Term Shutter
SUM	Software User Manual
SW	Software
SWIR	Shortwave Infrared
SWT	Science Working Team
TBC	To Be Continued ...
TBD	To Be Defined
TBS	To Be Specified

A Abbreviations

TBW	To Be Written
TC	TeleCommand
TCM	Trajectory Control Manoeuvre
TCP/IP	Transmission Control Protocol/Internet Protocol
TCR	Telemetry, Command and Ranging
TCS	Thermal Control Subsystem
TDMA	Time Division Multiple Access
TDRS	Tracking and Data Relay Satellite
THM	THerMistor
TID	Total Ionisation Dose
TIR	Thermal Infrared
TIR	Thermal Imaging Radiometer
TIS	Thermal Imaging Spectrometer
TJ	Triple Junction
TM	Telemetry
TM/TC	Telemetry / Telecommand
TMA	Three Mirror Anastigmat
TMM	Thermal Matematical Model
TN	Technical Note
TNS	Thermal Neutron Sensor
TO	Transfer Orbit
TOF	Time Of Flight
TPL	Acceptance Test Plan
TR	Transient Recorder
TRL	Technology Readiness Level
TRM	Test Readiness Meeting
TRR	Test Readiness Review
TS	Timing System
TT&C	Telemetry, Tracking and Control
TT&R	Telecommand, Telemetry and Ranging
TV	Thermal Vacuum
TVC	Thermal Vacuum Chamber
TWTA	Travelling Wave Tube Amplifier
TX	Transmitter
Tx	Transmit
TX/RX	Transmitter/Receiver

U/L	UpLink
UART	Universal Asynchronous Receive and Transmit
UHF	Ultra High Frequency
USMI	Ultraviolet Spectral Mapping Instrument
USO	Ultra-stable Oscillator
UTC	Universal Time Coordinated
VCDU	Virtual Channel Data Unit
vH&S	von Hoerner & Sulger GmbH
VIS	VISible wavelength range
VIS-NIR	VIS-NIR Mapping Spectrometer
V-Modell	Vorgehensmodell (has the sense of how to proceed)
VNIR	Visual and Near-Infrared
VSWR	Voltage Standing Wave Ratio
w/o	without
WBS	Work Breakdown Structure
WCA	Worst Case Analysis
WHM	WeilHeiM
WP	Work Package
WPD	Work Package Description
WWW	World Wide Web
X-FE	X-Band Front End
XML	EXtensible Markup Language
XPD	Cross Polar Discrimination
XRF-L	Lunar X-ray Fluorescence Experiment
XSD	XML Schema Definition