Technology Driven Robotic - Moon - Mission 2016

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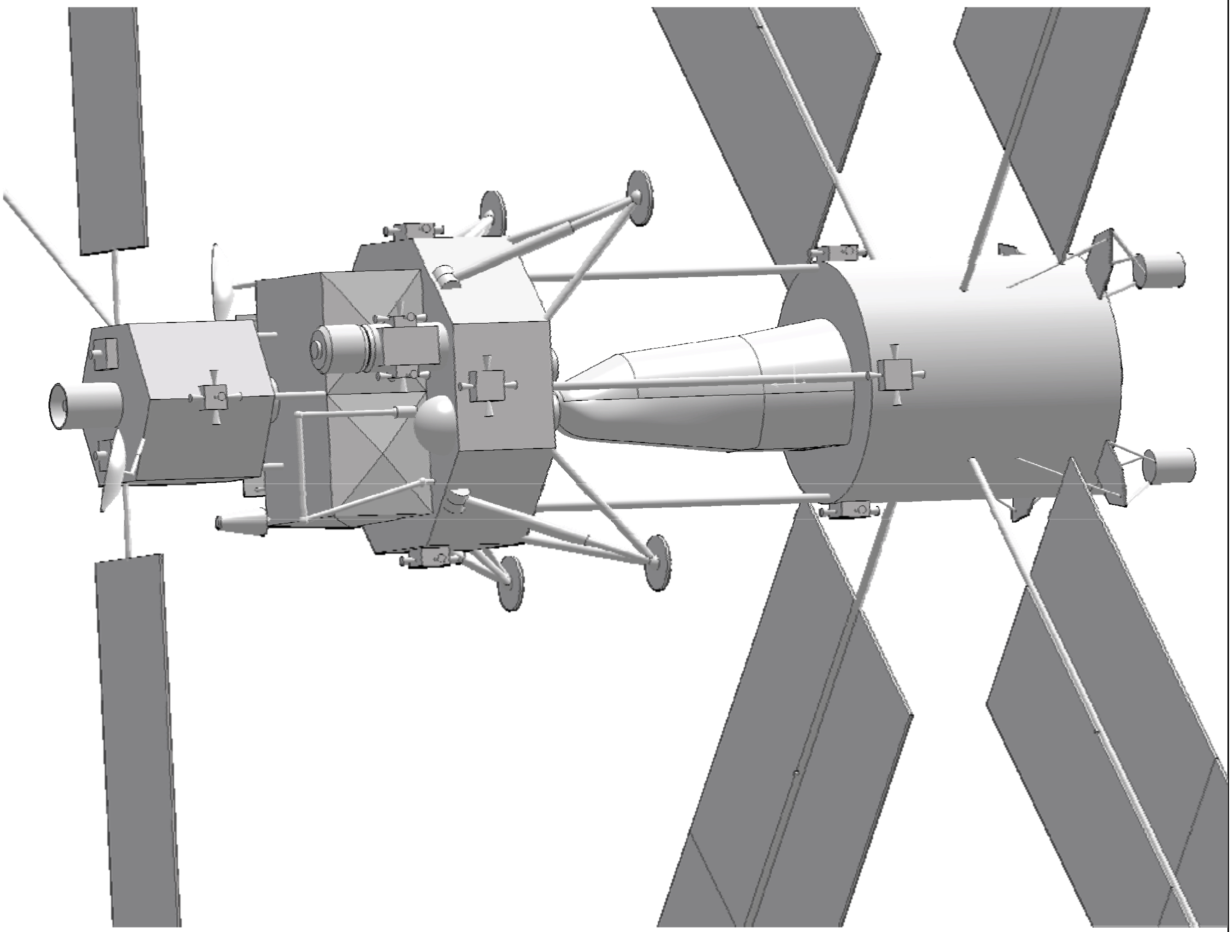

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Summary

The paper proposes a concept mission to Moon including a space-tug-vehicle in Moon orbit, a transfer surveillance/relay satellite into low lunar orbit, a Moon lander equipped with a rover for miscellaneous challenges and an Earth return spacecraft transporting Moon samples. To guaranty a low mission cost, trajectories of low impulse has been selected in combination of technologies like combined chemical-electrical propulsion; broad Ka–band/ X–band/ S-band transponder communication system and advanced solar arrays. The present analysis includes first global mass budgets for the lunar-space-tug-vehicle and its component modules and energy budgets for subsystems like the electric power supply and propellants. Also, here are defined the propulsion properties for the different modules.

1. Introduction

AURORA is ESA' long term technology program devoted to prepare human exploration mission to Mars. To achieve this goal, it is necessary previously to acquire knowledge and to improve the existing European technology level through robotic missions, where the Moon appears as an attractive platform for such tests. In particular, the German Aerospace Center, DLR, as the largest engineer center in Germany with a primary role to develop aerospace technologies, is motivated to be prepared for the AURORA vision. In the year 2006 the DLR Institute of Aerodynamics and Flow Technology located in Braunschweig/Göttingen has realized a feasibility study for a Moon sample return mission comprising a palette of technologies defined in ESA' technology matrix for AURORA. Some of the technologies to be mastered for future interplanetary missions are advanced propulsion systems; power subsystems; guidance and navigation systems; dynamic communication grid; intelligent control of vehicle components; systems for remote sensing and image mapping; rendezvous-docking systems; super orbital Earth re-entry with or w/o aero-braking, among others. The present paper discuses a flexible concept which enables to prove and qualify different new key technologies at a reasonable cost, while satisfying also scientific goals [1-2].

2. Mission Outflow

The space tug vehicle (**Fig. 1**) comprises a lunar polar-orbit satellite for relay communications and potential remote scientific investigations; a Moon lander with a rover for surface automatic research and a lunar ascent stage module for transportation probes from the Moon surface to a low lunar orbit; finally a return spacecraft for transportation of the probes to Earth. The required power is ensured by a high effective solar arrays mounted partly on the lunar polar orbit satellite but mostly on the Moon exploration and recovery vehicle. All along the journey, the solar arrays are interconnected to deliver the required electric power. The present study comprises a reference mission with an alternative respect Earth escape and Earth recovery scenarios.

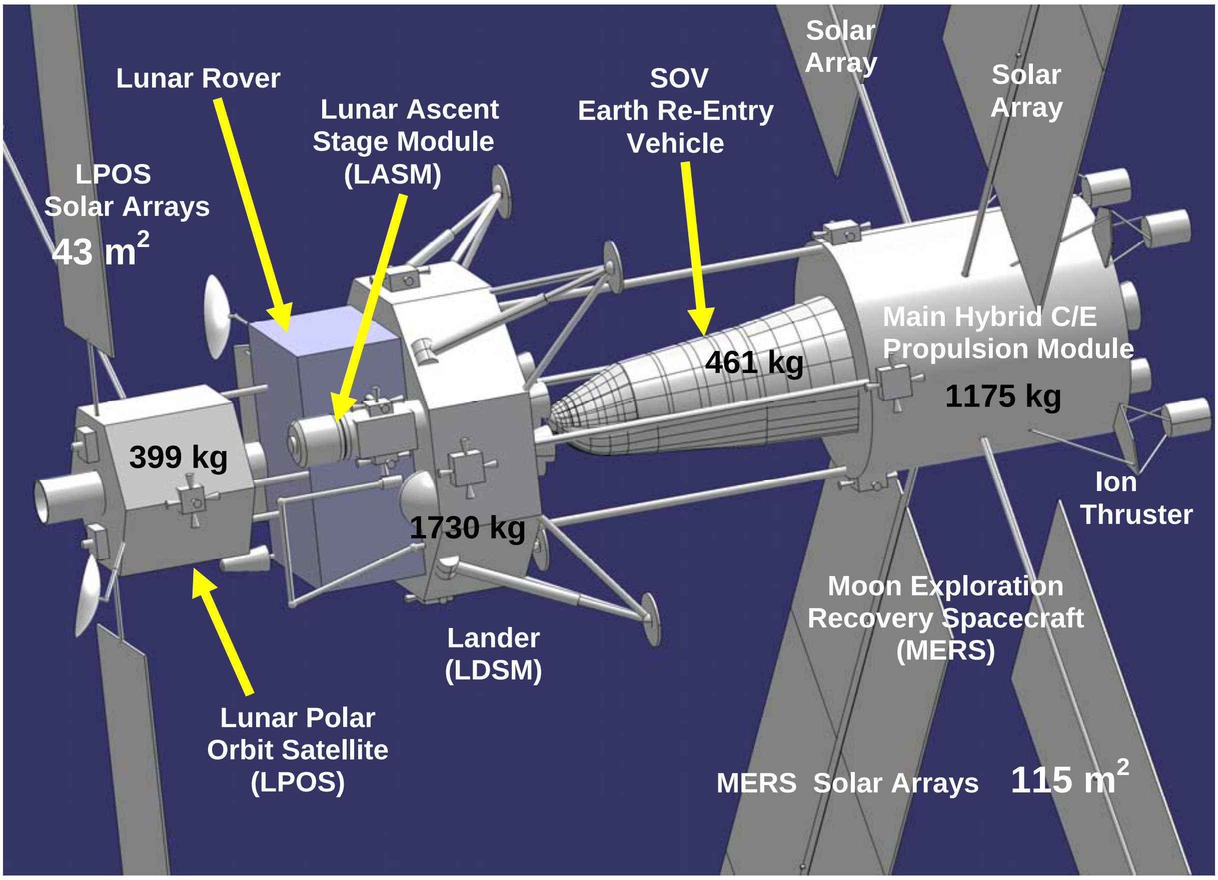


Fig. 1: Space Tug Configuration

The mission is scheduled to be started aboard an Ariane 5 rocket launcher (**Fig. 2**). Alternative could be considered launchers like the Angara or Proton. The space tug will be first inserted in nearly circular low Earth orbit (LEO) at 300km altitude. From this orbit and with help of the A5 cryogenic upper stage the space tug will be transferred to Geostationary Orbit (GEO). From GEO, the journey will continue by means of 4 ion-engines bonded in a cluster. In a time frame of about 11 months, the engines delivering a permanently small but almost constant thrust will move the space tug along a spiral escape trajectory up to the lunar transfer orbit [3]. The capture in the lunar orbit will be performed by passing in vicinity of the libration point L2, behind the Moon. L1 to L4 are Lagrangian points where the Earth and Moon attraction forces are in equilibrium. In the vicinity of the libration point L2, the trajectory will be corrected and the space tug reverse positioned regarding flight direction. From the libration point L2 the space tug will be decelerated by small successive thrust brake till it will be captured in a spiral entry trajectory and by arriving to the Moon parked in an approx. circular polar low lunar orbit with periselenium of 100km (**Fig. 3**).

After parked in the low lunar orbit, the polar-orbit satellite will be disconnected from the space tug and transferred to one, still lower, lunar orbit with periselenium of 50km. This orbit provides the best opportunity for in-depth investigation of the Moon surface, particularly the landing site. Once the polar-orbit satellite selects the landing site, the lunar descent stage module carrying the rover and the ascent stage module will separate from the recovery spacecraft and travel to a 100km x 20km elliptical orbit. The new orbit could be so adjusted that the lander will fly over the landing site in two or three passing orbits with periselenium of 20km, before the landing process be authorized [4]. Short before landing, some hovering may take place, enabling the lander to avoid some obstacles and finalizing with a soft landing. Once on Moon surface, the rover will be activated for the mission, moving out of the lander.

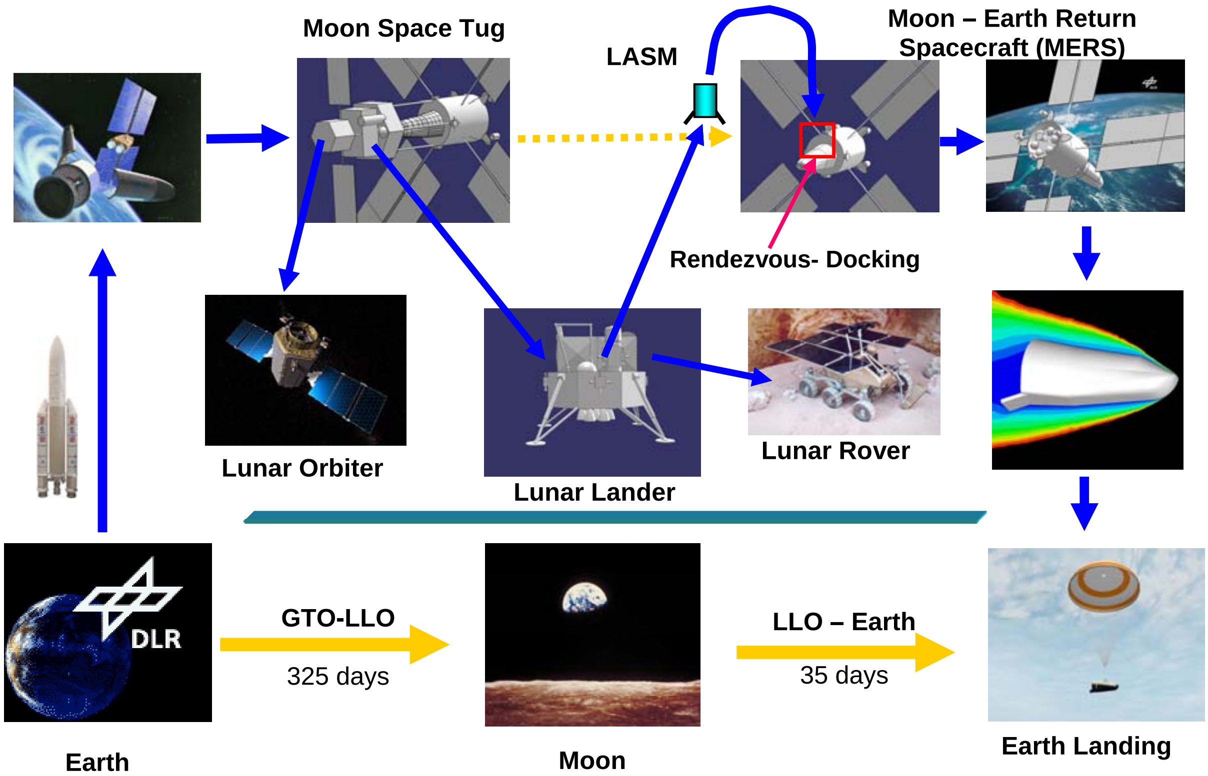


Fig. 2: Mission outflow

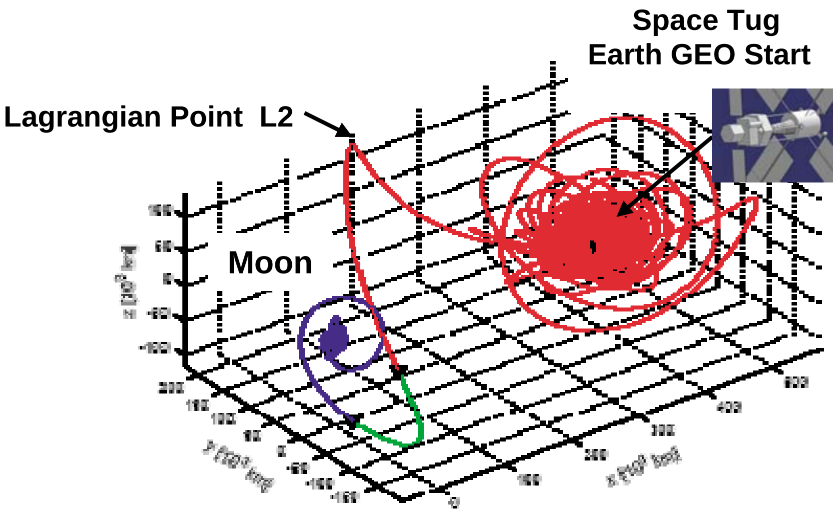


Fig. 3: Earth-escape trajectory and Moon-capture trajectory displayed in rotating selenocentric frame

After collect representative lunar probes, the rover will carry them to the lander and reload into the lunar ascent module. The ascent module uses the lander as launching platform. After a successful rendezvous-docking maneuver, the samples will be transferred to the recovery vehicle. Finally the ascent module will be ejected and the recovery module will initiate the return journey direction Earth. The recovery stage will depart from the Moon using electric propulsion in an escape-spiral low-thrust trajectory. The return journey is estimated in about 35 days. A combination of electrical propulsion and a chemical reaction control system will be used to inject the vehicle in Earth transfer orbit and from there to a further sustained travel to the Earth. Before re-entry in the atmosphere the main propulsion module will be ejected while the recovery vehicle with the payload will initiate a direct entry in Earth atmosphere at more than 11.3 km/s, terminating with a safety landing.

2.1 Mission option

The reference mission presents the lowest start mass and is the most promising in term of performance since the flying time toward the Moon is the shortest of all potential missions considered but the corresponding costs are the highest. On the other extreme, a possible scenario to realize is to start the journey from LEO instead GEO. For this option the start mass required is almost 500kg more that the reference mission and the estimated time to travel to the Moon around 16 months, i.e. 4 to 5 month longer than the reference mission while that is the time needed to goes from LEO to GEO. But the mission could be accomplishing with a less powerful launcher and hence the mission costs are the lowest. A risk in this mission is the longer residence time within two radiation belts due to the duration of the trip between LEO and GEO which may produce a loss in solar array efficiency up to 10%.

3. Space Tug Modules

3.1 Lunar Polar-Orbit Satellite

It is foreseen a spacecraft with a payload capability up to 40 kg (**Fig. 4**), which may house diverse scientific equipments but for the mission it shall provide support for landing operations, i.e. search and identify for the place and preparation for the touch-down maneuver and be the communication link between the rover and Earth. Communications are planed on a triple band transponder system. Broad Ka-band (32-34 GHz, 500kBit/s) will be used for communication between Earth and Moon, particularly for teleoperations and global telemetry. X-band transponders (7-8 GHz) will be used primary for the communications with the rover, like telemetry and tracking. While an S-band transponder will be reserved for communications with the lander and emergency communication link with the rover via the lander.

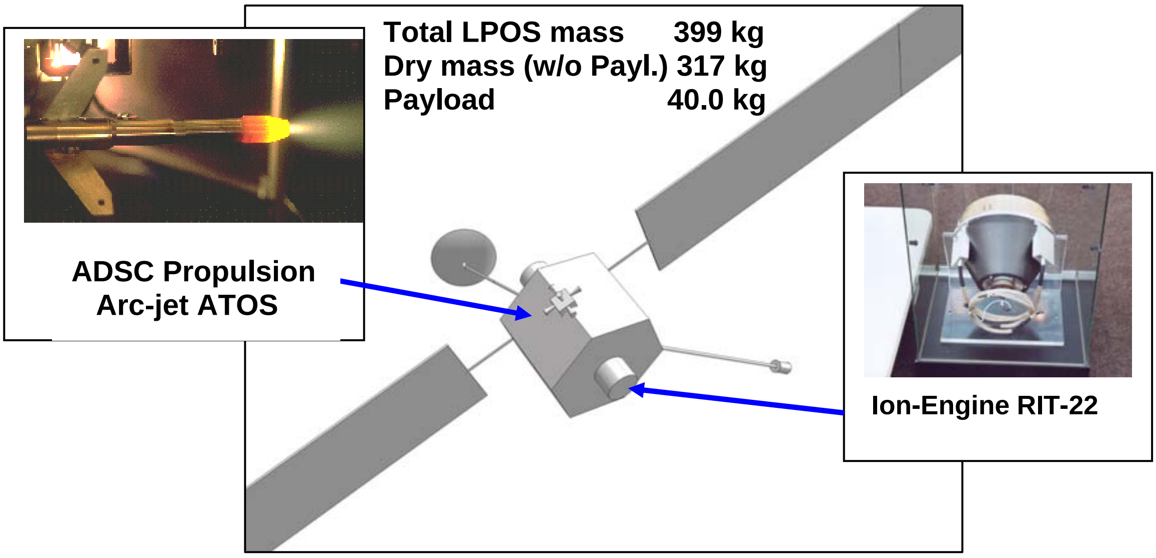


Fig. 4: Lunar Polar-Orbit Satellite (LPOS)

The total mass estimated for the satellite including propellant is 399kg and the dry mass with out payload is 317kg. It has a volume of approx. 1m^3 and a hexagonal cylinder form, enabling to be stowed under the launcher fairing together with its solar arrays with a surface of 43m^2.

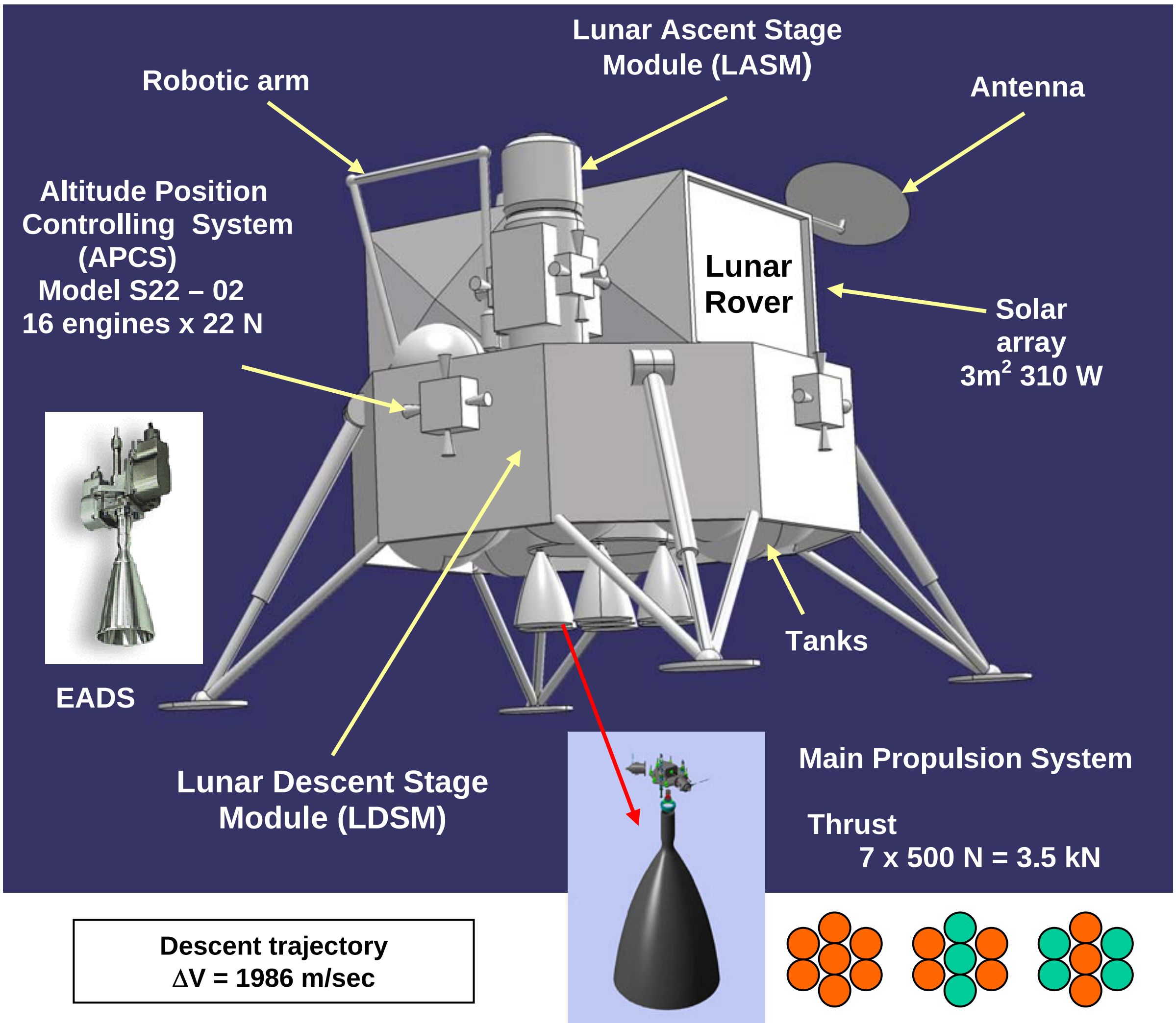
The necessary power budget in full operating mode is 6700W. Lithium-ion batteries with specific energy density of 144W-hr/kg are planed as power supply backup in case a lunar eclipse and/or in emergency. The navigation system includes an on board autonomous navigation/ data handling system conceptually similar to that of the SMART-1 spacecraft. A sun sensor and a star mapped will provide information about position and altitude. Fiber optic laser gyros are foreseen for safe modes and rate dumping. To enable accurate 3-axis stabilized attitude control, the altitude determination and control system includes three reaction wheels actuators combined with low cost arc-jet thrusters as reaction control system. The selected propulsion system for attitude control is an arc-jet type. The arc-jet thruster has a specific impulse 60% higher than a corresponding chemical engine, lower mass flow and pretty lower contamination of the solar arrays. Here selected is the arc-jet ATOS, of the University Stuttgart, with a specific impulse of 480s, 115mN thrust and nominal electric input of 750W. For trajectory correction and trajectory transfer a radio-frequency ion thruster RIT-22 of EADS, is selected. A small xenon gas mass flow with high specific impulse, more than 6000s, enables a higher life cycle of the satellite in Moon orbit.

3.2 Lander Spacecraft

The lander spacecraft consists of three modules: the descent module (**Fig. 5**), the rover (**Fig. 6**) and the ascent module (**Fig. 7a**). The total mass inclusive payload is estimated about 1730kg while the mass to be transported to the Moon surface, dry mass, 937kg (certainly, the corresponding weight due to Moon gravitation will be approximately six times lower). These masses are calculated assuming the landing will take place at Moon latitudes between 60° - 80°. For a hypothetical mission to the South or North Pole regions, these masses must be increased.

3.2.1 Descent Stage Module

The primary objective of this module is a soft landing on the lunar surface and delivers the lunar rover and the ascent module. Also the lander will provide with emergency link between rover and Earth in case that the orbital satellite is not in the range of the rover transponders; supervises the deploy and motion of the rover and transfers with a robotic arm the soil probes from the rover to the ascent module. The proposed design has an octahedral form with four pneumatically amortized legs able to withstand acceleration up $50m/s^2$. The equipment will include a main descent propulsion system, position control system, propulsion system, power supply (solar arrays, lithium-ion batteries and battery management system), avionics (sensors, on board autonomous navigation / data handling system, telemetry and tracking, communication equipment) and a deployable landing bridge for the rover. The minimum operational life time for the spacecraft on the Moon surface will be 6 lunar days. The power supply is guarantee by solar arrays with overall surface of $5m^2$ and backup batteries compatible with those installed in the lunar polar-orbit satellite and in the recovery stage. A multi link S-band transponder (2076 MHz/ 2255 MHz) will enable direct communication with Earth ground-stations or via the polar-orbit satellite. However, the first method, i.e. direct communication, is preferred since the data uplink and downlink could be up to 70% faster than using the Moon satellite for communication links. The maximum thrust requirements for the main descent propulsion system is 3500N. A cluster of 7 European Apogee Motors (EAM) providing 335s of specific impulse and 500N thrust each (and up to 600N for less than 12s) is selected. A thrust-throttling with this solution is possible using a step mode, i.e. alternatively action of 1-3-4 or 7 engines. The position control system consists of 16 bipropellant rocket engines S22-02 of EADS. This engine has a mass of 690g, provides a thrust in vacuum of 22N and a specific impulse of 290s.



European Apogee Motor (EAM - EADS)

Fig. 5: Lander Spacecraft

3.2.2 Lunar Rover

The present analysis does not intent to provide with data for a rover but to provide the baseline envelop for its design. According, here is estimated only the minimum mass necessary for a robust durable surface vehicle that within a life cycle of 6 to 12 lunar days be capable to overcome the difficulties of an unstructured terrain and capable to collect soil probes in a radius of 40km from the landing point. A total mass of 300kg is estimated where 30kg are foreseen for scientific payloads. The specific scientific profile is not a matter of this study and shall be determined from specialists for lunar sciences. The necessary power supply calculated for full operation mode is 465W. A rover powered only with solar arrays would impose limitations to the landing place selection and also to mission durability since solar arrays due to radiation and exposition to the Moon dust, degrade within few years. Here a combination of state of art solar arrays with high technology batteries is recommended [5].

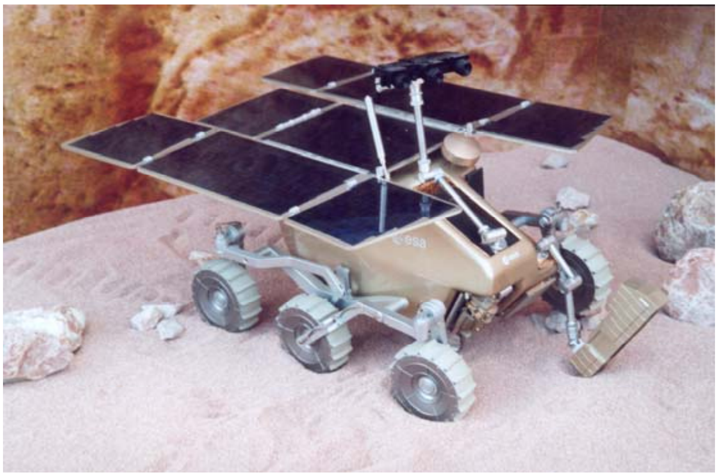


Fig. 6: Lunar Rover

3.2.3 Ascent Stage Module

The ascent module shall transport the container with the soil-samples from the Moon surface to the return vehicle parked in circular lunar low orbit and hence its major technological task is a rendezvous and docking maneuver. The ascent module will be equipped therefore with one miniaturized inertial guidance and control systems which includes the propulsion thrust vectoring for the ascent. As main thruster a European Apogee Motor, already described in chapter 3.2.1 is selected, while the reaction control system (RCS) consists of 10 bipropellant rocket engines S10-02 of EADS, with a thrust in vacuum of 10N, specific impulse 286s and 14kW power. A rendezvous-docking maneuver in lunar orbit is a challenging maneuver because the time delay in the communications, up to 2s, will impact the accuracy requested for designing the reaction control system. Also the navigation must be highly precise. The automated rendezvous-docking system may be based on present EADS dual system, but miniaturized (**Fig. 7b**). This approach comprises a system based on relative GPS technique, produced by Matra Marconi Space, for the control of the long range rendezvous trajectory part (beginning after the departure from the lunar surface and ending at 250m from the return vehicle) and a laser rendezvous sensor system produced by DASA Jena Optronics, for the final short range guidance from 250m up to the docking.

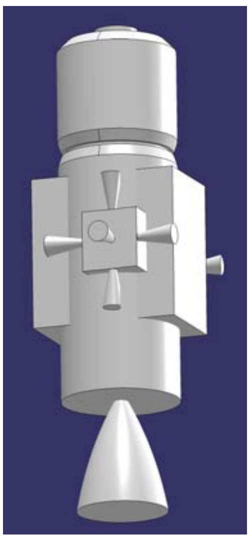


Fig. 7a: Ascent Stage Module

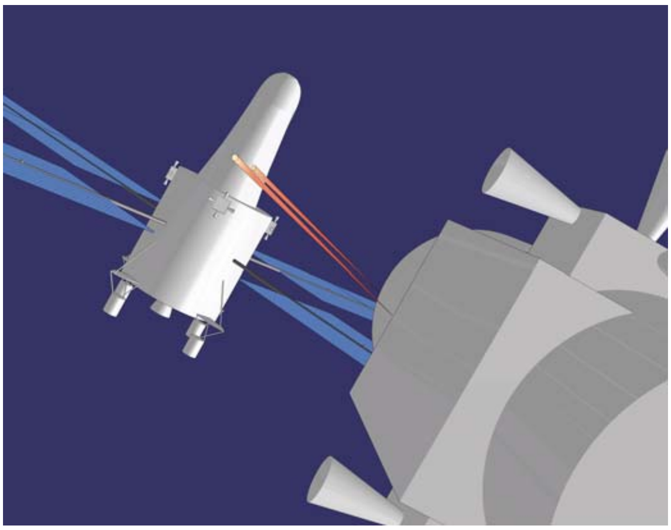


Fig. 7b: Artist view of the rendezvous maneuver

3.3 Return Spacecraft

Figure 8 shows the return spacecraft consisting of the main hybrid propulsion module and the Earth re-entry vehicle. The total spacecraft mass is calculated on 1640 kg.

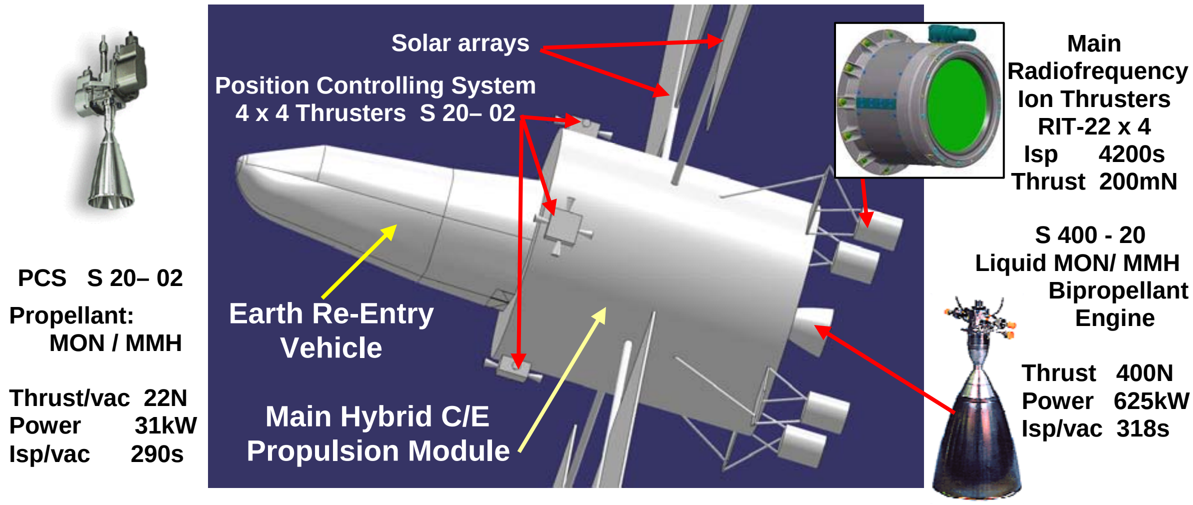


Fig. 8: Return Spacecraft

3.3.1 Main Hybrid Propulsion Module

The main hybrid propulsion module serve as carrier for the earth re-entry vehicle, comprising a combined chemical-electrical propulsion system, an S-band transponder enabling communication and low-resolution image transfer necessary for the control of operations like lunar descent module separation and Earth re-entry vehicle separation; a thermal controlling system and a pyrotechnic separation mechanism for the return vehicle. Additionally, the module is equipped with a position control system comprising 16 liquid propulsion thrusters for course corrections. The dry mass of the module is estimated on 527 kg.

The exposed concept intends the application of electric propulsion along the marching route Earth-Moon-Earth trajectory. The application of liquid propulsion is planed only for course corrections like final lunar low orbit approach, trajectory approach by rendezvous maneuver with the ascent module and Earth recovery maneuver, among others. The operation time for the liquid propulsion must be limited to maintain a high value for the average specific impulse of the complete propulsion system along the whole mission. Within this concept the burning time for the liquid propulsion thrusters is limited to 12.3% of the overall ΔV budget; the estimated delta velocity budget for this purpose is approx. 700m/s. Furthermore, 4 Radiofrequency Ion Thrusters of the type RIT-22 [6] are selected for the electric propulsion system since they are unique in providing a very high value of specific impulse, almost 7000s each, high electric efficiency, above 90% with a total efficiency higher than 80%. In the standard design the mass of a RIT doesn't exceed 18kg and the corresponding power consumption lies between 3.2kW to 6kW, based on a test with positive voltage about 1.5kV. For higher voltages the power consumption and thruster mass increase moderate. 4 RIT-22 engines are needed to enable permanently up to 0.8N thrust and for short term up to 1N thrust. The engines will be mounted on a redundant gimballing mechanism which enables direction control during the cruise flight. As main thruster, the EADS S400-20 [7] is selected. It develops 400N nominal thrust in vacuum with a specific impulse of 318s and presents a low mass of 3.6kg and compact dimension. This bipropellant thruster is a very reliable engine for long duration mission as it demonstrated in a great number of satellites/ spacecrafts like the series Intelsat, Meteosat, Astra, Rosseta, Mars-Express etc.

The primary power is based on solar arrays providing a maximal power load of 24700W. Two concepts are considered: Stretched Lens Array Square Rigger Technology with solar concentrators [8] and Multi-junction GaInP/GaAs/Ge solar cells with flexible composite panel design/ mast deployable arrays. For the present analysis a solar illumination intensity of 1333.6 W/m^2 when the Moon is in opposition to the Earth and Sun (higher distance) is considered. The calculation takes also in account the solar panel efficiency, inherent degradation due design inefficiency, temperature degradation and array shadowing. The stretched lens array technology has higher panel efficiency, 23% at operating temperature, higher power density (> 300 W/m^2) and lower total surface, 82 m^2 compared to 158 m^2 for the multi-junction GaInP/GaAs/Ge solar cells. But due to the additional concentrators it has higher total mass for the same load. Since total mass is the critical parameter for the mission, the last option, i.e. the Multi-junction GaInP/GaAs/Ge solar cells technology is selected due to a lower total mass, 234 kg, and simpler construction. While 72 % of the total solar-panel surface will be allocated by means of 4 arrays on the main propulsion module and the rest in two arrays mounted on the lunar polar-orbit satellite, during the journey to the Moon all 6 arrays will be interconnected and the power directed to the four RIT-22 thrusters.

3.3.2 Earth Re-entry Vehicle

The re-entry vehicle will transfer the lunar samples to the Earth demonstrating an atmospheric re-entry at velocities over 11 km/s. Planed is an actively controlled vehicle which may maneuver autonomously to improve landed location. Within this goal it is important to qualify previously the necessary advanced thermal protection systems and hot structures under representative flight conditions and a flight control system based on aerodynamic surfaces and center of gravity displacement. Three classes of vehicle are possible: capsule, lifting body and winged body. The advantage of the capsule concept is that it features a simple design but presents a very low design evolution capability; its gliding capacity is very low; has limited guidance and controlling properties and the landing range is broad, resulting on larger costs for recovery operations. A winged body presents a considerably higher mass compared to other concept classes and therefore is not a feasible concept for a Moon mission. Also the protection of the wings from the resulting re-entry heating due to a direct re-entry with super orbital velocity is hardly problem. Here the lifting body class has been selected as the best option. The selected concept has a good design feasibility, robustness, maturity and growth potential. This selection is fully compatible with extensive national efforts to develop a fully controllable re-entry vehicle, like the German DLR SHEFEX Program and the French CNES PRE-X Project. Up to day the interplanetary missions performed by the USA, Japan and Russia use simple ballistic capsule forms like sphere, cone etc.

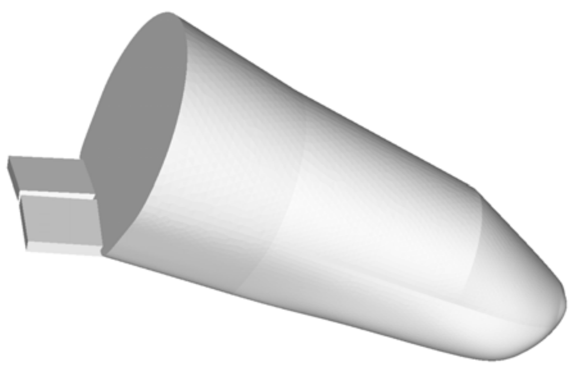


Fig. 9: Earth Re-Entry Vehicle

The design of the shape ensures maximum volume efficiency due to a limited weight of 460 kg. The interior equipment include the docking system for lunar samples container; avionics like internal sensors and the telemetry tracking system; Li-Ion battery supply; actuators system for flaps and parachute/safeguard system as well as place for potential passenger experiments; flashing rendezvous beacon/retractable forward-facing spot lights and airbags for a soft landing. For the present study a lifting body aeroshape having a length of 2.2m, a width of 1.1m and height of 1m is selected (**Fig. 9**). The nose has a moderated radius to enable that the nose cap material withstands the harsh environment under re-entry conditions. The selected shape exhibits a lift to drag ratio in the range 0.6 to 0.9. Two body flaps, main components of the aerodynamic control system, are located behind the vehicle in a cantilevered configuration. At the same time forward longitudinal and downward vertical centering to assure conditions for longitudinal/lateral stability is foreseen. The selected shape is a generic form and will be optimized within further studies, being the extreme heat loads during re-entry and the stability, controllability and trimmability of the vehicle the primary goals of such activities. The aerothermal loads are dependent of the re-entry trajectory-profile selected for the mission (**Fig. 10**). Principally it is possible to perform an Earth re-entry from a hyperbolic trajectory directly with super orbital velocity or by previous reduction of the velocity from second cosmic velocity to first cosmic velocity by means an aero-braking maneuver. Aero-braking is the term applied to the practice of change the spacecraft orbit by using the atmospheric drag to reduce the orbit energy in repeated passes. It is a concept that significantly reduces the necessary propellant for orbit insertion and consequences the launch mass and therefore launches cost. However, the design of a fully automated aero-braking maneuver requires a detailed interaction between navigation and spacecraft team; sequencing check, mission management and atmosphere advisory group. Today, in spite of the advances in aero-braking automation, it remains a human-intensive and costly process. Here only heat loads resulting from a direct reentry are considered. Performing a direct super orbital velocity re-entry, the thermal protection system will be non reusable ablative but the technological requirements are lower than those in case of a re-entry accomplish with an aero-braking maneuver and the mission risks and costs for guidance and controls also lower. For a flight path angle lower than 10deg the heat flux at the stagnation point of the vehicle nose cap is about 14 MW/m^2. A possible solution for the thermal protection system may be based on carbon phenolic ablator composite materials [9]. Furthermore, the maximal vehicle deceleration may not overshoot the 50g to allow the use of aluminum alloy for the primary structure.

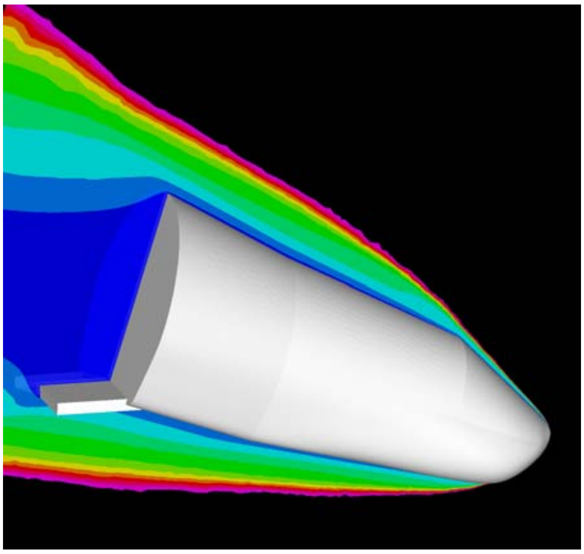


Fig. 10: Contours of constant Mach for a re-entry at Mach 25, 79km altitude

4. Conclusion

A feasibility study for a robotic mission to the Moon has been presented and discussed. It is a sample return mission which includes a lander module carrying a rover to investigate the Moon surface area within a radius of 40km from the landing position; collecting representative lunar soil probes and bringing them back to Earth.

A number of new technologies necessary for the mission have been identified like the selection of low impulse trajectories; hybrid chemical-electrical propulsion with average specific impulse over 2200s; arc-jet propulsion for the lunar satellite attitude control; advanced multi-junction solar arrays for primary energy supply; automated semi-autonomous rendezvous-docking system; broad Ka–band/ X–band communication system for data transmission; precise guidance during trans-lunar trajectory to enable orbital insertion; precise guidance for targeting landing position on the Moon; super orbital velocity Earth re-entry and recovery maneuver.

The study indicates the proposed mission is feasible despite the enormous challenge it comprises. However, further deeper studies should follow to collect experience and validate some critical technologies as like precise space-navigation; rendezvous maneuver in lunar orbit and the reentry with second cosmic velocity.

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