The SMART-1 Mission

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The SMART-1 mission, to be launched at the end of 2001, is intended to demonstrate innovative and key technologies for deep-space scientific missions. Its use, for example, of solar electric propulsion as its primary drive mechanism will be a first for Europe and is essential in paving the way for future ESA projects with large velocity requirements, such as the Mercury Cornerstone mission. SMART-1 will also be a test case for a new approach in terms of implementation strategy and spacecraft procurement for the ESA Science Programme.

The total life-cost budget allocated to SMART-1 is 50 MECU. This budget constraint imposes use of a cheap launch option, such as an Ariane-5 auxiliary payload launch into a standard GTO or a Rockot escape-trajectory launch. This in turn limits the planetary bodies that can be reached within a given short (1.5 - 2 year) overall mission lifetime, which do, however, include the Moon and Earth-crossing asteroids or comets.

The mission is presently under Phase-B definition by the Swedish Space Corporation. The funding of the mission is being used to compensate Sweden and Switzerland for the deficits in their industrial returns. The mission is expected to be funded partly by France and the United Kingdom, which presently have an industrial-return surplus. The Directorate of Industrial Matters and Technology Programmes, via the Technology Research Programme, and the Directorate of Scientific Programmes would provide the remainder of the funding, within agreed limits. The mission would thus effectively be a partnership between ESA and the participating Member States.

Introduction

SMART-1 is the first of the Small Missions for Advanced Research in Technology within ESA's Mandatory Scientific Programme (Fig. 1). These missions have been introduced by the Agency as one of the strategic elements for reintroducing balance and flexibility into the Horizons 2000 Science Plan. They constitute a preparatory technology-development programme focusing on items identified as critical to the success of the Cornerstone missions, including flight demonstrations where deemed appropriate. The scientific importance of the SMART-1 mission therefore resides mainly in its preparatory nature for upcoming scientific missions, and in particular for those missions that will benefit from primary electric propulsion and deep-space communications.

The importance of Solar Electric Primary Propulsion (SEPP), i.e. electric propulsion fed by Sun-generated electrical power and used as the spacecraft's main propulsion system, has been well-recognised in several past studies, including the Mercury Cornerstone study (to enable a low-circular-orbit mission). Earlier studies of the Solar Corona Probe and Solar Stereo missions had also identified SEPP as being of primary importance. Astronomy interferometric missions like the IR Interferometer Cornerstone and LISA might also benefit from the timely development of such a technology.

SMART-1 will therefore demonstrate the use of SEPP on a small mission representative of future deep-space scientific mission, with the emphasis on the common system aspects, rather than the choice of a particular engine, which is more mission-specific. Several other

Table 1. Typical specific impulse (Isp) and thrust levels for satellite on-board propulsion systems

	Typical Chemical Propulsion			Electric Propulsion		
Engine	Solid Boost Motor	Hydraxine mono-prop.	Hydraxine bi-prop.	PPS-1350	RIT-10	UK-10
Isp [s]	290	220	300	1600	3000	3200
Thrust [N]	50 000	0.5 to 400	4 to 6 000	0.070	0.020	0.020

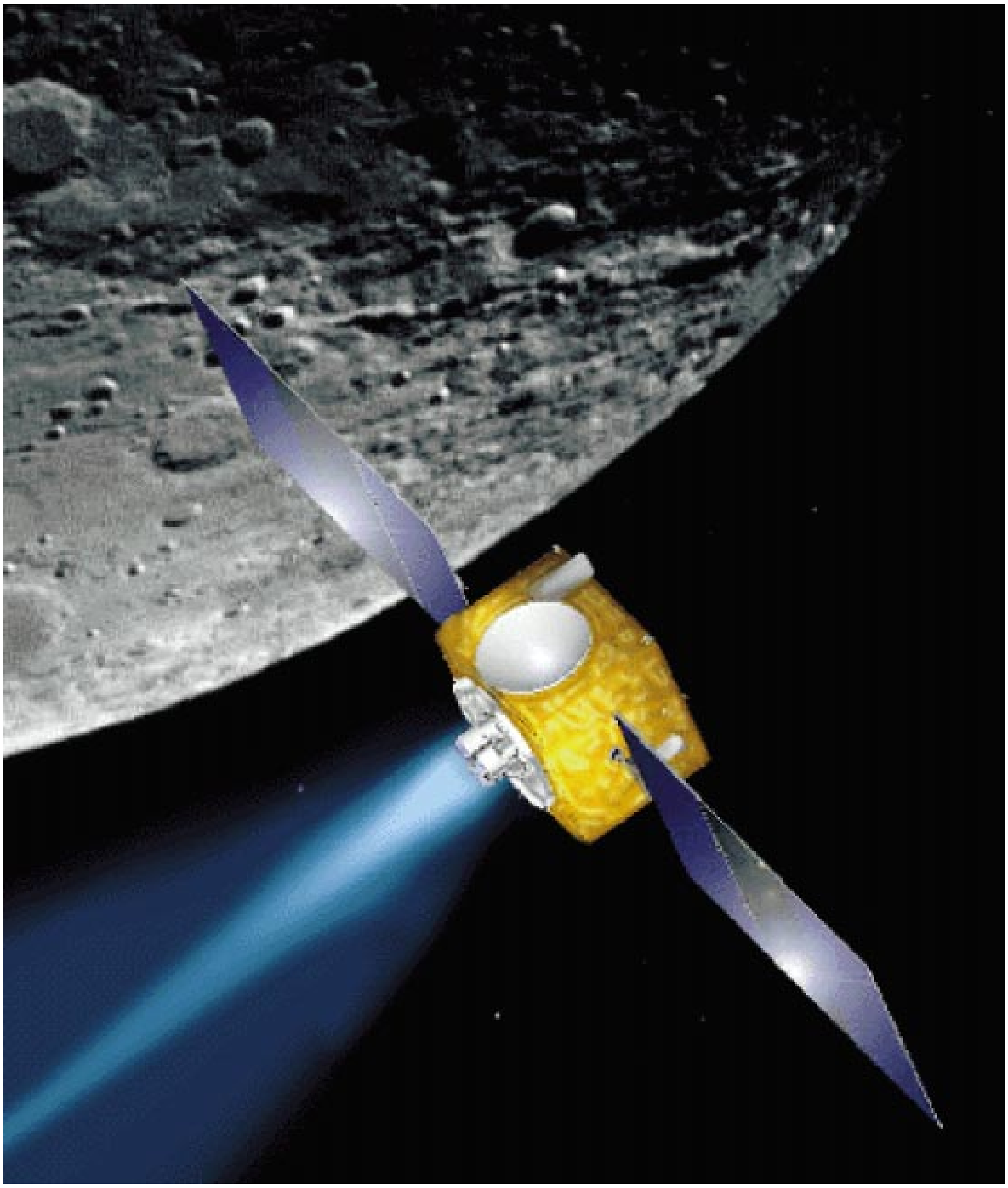

Figure 1. Artist's impression of the SMART-1 spacecraft

technologies are planned to be tested, including some of the candidate technologies that pave the way for approved Horizon 2000 scientific missions and others that are more general in nature (e.g. Li-C batteries, Ka-band, cascade solar cells, etc.) and would be useful for a wider range of future missions.

Being part of the Science Programme, it is also important that SMART-1 should achieve a valuable scientific return. An Announcement of Opportunity has therefore been issued for scientific instruments to be flown on this mission with a view to: (i) directly demonstrating the adequacy of the technology for science, and (ii) providing the scientific community with an early possibility for scientific investigation.

Mission overview

The low overall mission budget for SMART-1 means that a low-cost launch is required. An obvious European choice is therefore accommodation on an Ariane-5 as an auxiliary passenger. This, however, limits the spacecraft mass to 80 kg in the case of the ASAP-V platform for micro-piggybacking, or about 350 kg for a Cyclade configuration. In both cases the spacecraft would be delivered into a standard Geostationary Transfer Orbit (GTO) (perigee altitude 620 km, apogee altitude 35 946 km, inclination 7°, argument of perigee 178°, longitude of descending node 10° W).

Another alternative would be direct injection into an escape orbit by a small launcher such as the DASA/Khrunichev Rockot, which can deliver a 1900 kg payload into a 200 km x 200 km Low Earth Orbit (LEO). From there a suitable solid-rocket-motor upper stage can accelerate a 350-400 kg spacecraft onto a parabolic escape trajectory from the Earth's gravitational influence (Fig. 2).

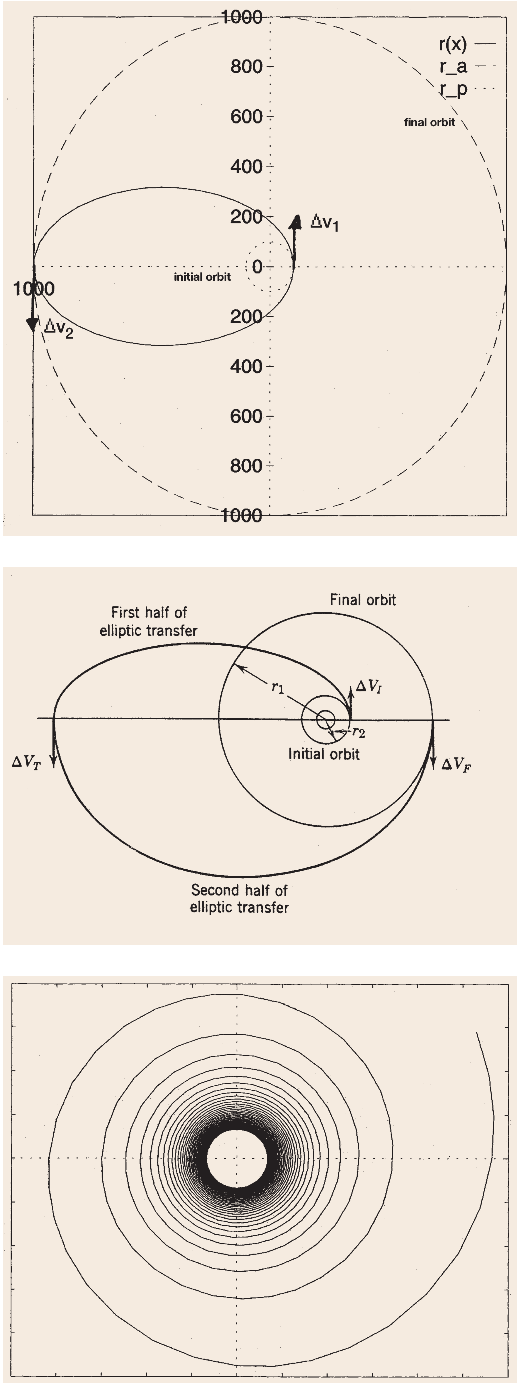

Figure 2. Comparison of impulsive (a and b) and low-thrust (c) orbit-transfer trajectories

a. Hohmann transfer

b. Bi-elliptical transfer

c. Low-thrust trajectory

With these two launch options, two types of planetary bodies generally classified as Near-Earth Objects (NEO) can be reached: the Moon and Earth-crossing asteroids or comets. Three mission options have therefore been considered for the preliminary assessment of the SMART-1 mission:

– A mission whose trajectory is bound to the Earth-Moon system. This includes missions to the Moon, with weak capture in an elliptical lunar orbit. Alternatively Earth-Moon system tours can be conceived involving flybys or rendezvous with the Moon or the equilateral L4/L5 Lagrangian points. The allowable payload mass varies from 10 to 20 kg and the minimum mission lifetime from 250 to 450 days.
– A flyby to an NEO, either an asteroid or a comet. This mission can be performed with an Ariane-5 launch to GTO as an auxiliary passenger. The payload mass is limited to maximum of 10 kg and the minimum mission lifetime exceeds 2.5 years.
– An NEO rendezvous mission using a dedicated launcher like Eurockot. The expected payload mass, depending on the chosen asteroids and launch date, is approximately 20 kg, with some limited growth possibility. The minimum mission lifetime is about 1.5 years and varies greatly according to the target.

A cruise phase during which the electric-propulsion engine is not operated is a feature of all of the above mission categories. The length of these cruise phases, lasting from 100 days to 1 year, depends strongly on the mission option and target selected.

Lunar mission from GTO

An important difference between the Rockot and Ariane-5 GTO launch strategies is the escape strategy. For the Rockot option, there is

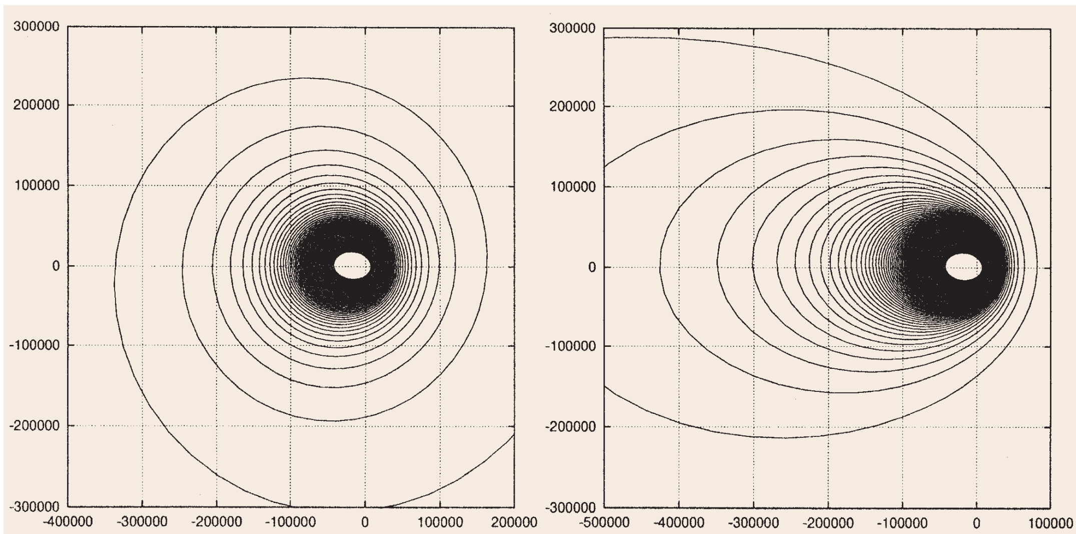

Figure 3. GTO escape spirals with (left) and without (right) coast arcs

sufficient mass available to allow an injection into lunar transfer using an additional boost stage. For the Ariane case, the spacecraft must spiral out, propelled by its SEPP thrust, spending a considerable time in the Earth's radiation belts. Solar arrays are very sensitive to damage from such radiation, which would result in a progressive degradation in SEPP performance. This solar-array degradation has therefore been modelled and coupled to the mission-analysis trajectory-optimisation software to provide direct information on the SEPP power available for a given trajectory. Typical GTO escape trajectories are shown in Figure 3.

The desired final lunar orbit is a polar 1000 km x 10 000 km elliptical orbit with its pericentre located at one of the poles. The optimisation of a low-thrust transfer trajectory connecting the GTO to such a lunar orbit is a difficult problem. Although several authors have tackled it and partially solved it for particular cases, no general optimisation procedure is yet available.

In the framework of the SMART-1 mission analysis, the problem has been tackled in several ways and the work is still in progress. The preliminary results indicate that the spiral out from GTO can be performed with either tangential or circumferential thrusting. The plane change for a year-long launch window can be accommodated by means of out-of-plane thrust. The introduction of coast arcs improves the trajectory for the engine with higher thrust. The capture in the lunar orbit is performed by passing in the vicinity of the cislunar L1 Lagrangian point. The low acceleration (in the order of 1×10^{-4} ms^{-2}) causes some problems with the capture and stability of the initial lunar orbit. Some results of the trajectory calculations are shown in Table 2, and a sample trajectory in Figure 4.

Table 2. Lunar mission trajectory performance

	SPT-100	UK-10 or RIT-10
ΔV	3.6 km/s	4.5 km/s
Fuel	64 kg (estimated)	33 kg
Payload mass	10 kg	24 kg
Time of flight	250 days	450 days

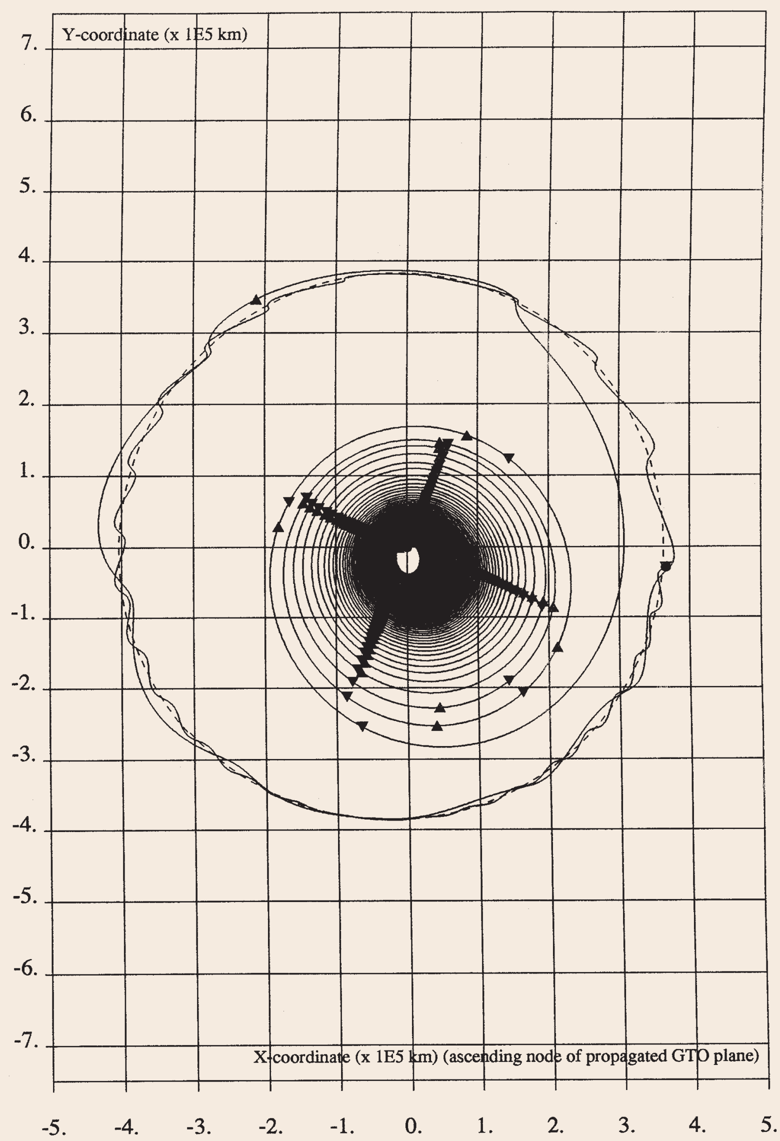

Figure 4. Lunar transfer trajectory in inertial coordinate system

Asteroid missions from GTO

NEO flyby and rendezvous have been studied starting from a GTO. First the escape spiral has to be performed, similar to that of the previous lunar transfer trajectory. In this case, however, the optimisation aims at minimising the fuel whilst maximising the orbital energy and no conditions for the lunar capture need to be set. The time required to achieve a parabolic escape varies between 230 and 450 days, depending on the type of engine and the thrust strategy used.

The target NEO has been selected from a Catalogue (ftp.lowell.edu) providing osculating elements of 35 065 asteroids at epoch September 1997, with launch assumed to take place in November 2001. Several trajectories have been computed with classical Pontryagin-type optimisation techniques, including a partial optimisation of the switching strategy. The results have shown that: (i) no asteroid rendezvous is feasible; (ii) just a few asteroids (1996 XB27, 1993 BX3, 1989 ML, etc.) and comets (Tempel-2, Haneda-Campos) can be reached, but only with high specific impulses (>3000 s); (iii) the flyby velocity is of the order of 10 km/s for the comets and 2-4 km/s for the asteroids; (iv) the minimum time of flight is of the order of 3 years. These results seem to discourage the choice of such a mission option.

Asteroid missions from direct injection
Clearly, an asteroid rendezvous mission offers the most promising scientific results. It also involves all of the typical features of a fully-fledged deep-space mission. The direct-injection launch option has been considered primarily for these reasons.

A mass at escape of 350 kg, with an optimised escape velocity and 20% gravity loss, is assumed for a launch date starting from November 2001. With these assumptions and using the high-specific-impulse engine (>2000s), many rendezvous opportunities with asteroids can be identified (Table 3). The "killer asteroid" 1997 XF11, for example, can be reached almost every year. This, however, is not the most favourable mission scenario.

Table 3. NEO rendezvous trajectory performance

Target	Launch date	Launch mass [kg]	Time of flight [days]	Payload mass [kg]
Orpheus	29-01-2002	321	442	42
1989 UQ	18-05-2002	320	416	14
1989 ML	25-07-2002	302	409	43
1993 HA	02-10-2002	317	518	25
1997 XF11	09-08-2001	310	1063	6
1997 XF11	29-07-2003	327	994	9

Table 4. Planetary science and instrumentation

Planetary Science	Instruments
Mass and gravimetry	Flyby tracking
Coarse volume and density	Micro-cameras
Rotational properties	Cameras
Coarse imaging/albedo	Narrow FOV imager
Geology and morphology	High-resolution camera
Stereo mapping / topography	High-resolution camera
Mineralogy	IR mapper
Geochemistry	X-ray spectro-imager
Planetary environment	Wide FOV and UV imager

The main problem with the asteroid missions investigated so far is the large distance from Earth at which the rendezvous takes place – up to 2 AU – placing high demands on the deep-space link. The mission-analysis effort is therefore continuing, seeking closer targets.

Science and technology objectives

Science objectives and model payload
We have seen that the bodies that could be explored in the course of this mission are Near-Earth Object(s) (NEO) and/or the Moon. The Moon is one of the oldest bodies of the Solar System and so, besides its own evolution, it has also recorded the first "footprints" of the Solar System's history. Its scientific study can therefore improve our understanding of the evolution of the Solar System, terrestrial planets, and the Earth-Moon system as well as of the Moon itself.

The NEOs form the present population of potential Earth impactors. Their exploration can provide insight into the physical nature of the bodies that have dominated our planet's cratering record since 4 Gyr and have apparently had an important impact on its biological evolution. It is generally agreed that there are two sources for these ephemeral bodies: (a) collisional fragments of main-belt asteroids delivered by efficient eccentricity pumping, due to resonances with Jupiter and Saturn, into Earth-approaching orbits; and (b) cometary nuclei surviving long past their active lifetime as inert objects, either devolatilized or covered by a surface crust of refractory material.

Some typical planetary-science goals, and the types of instruments on SMART-1 which could address them, are summarised in Table 4. Depending on the particular mission scenario chosen, the payload mass is between 10 and 25 kg. Important science – for instance involving astrophysical observations – can also be carried out during the mission's cruise phase, as indicated in Table 5.

Table 5. Cruise science and instrumentation

Cruise Science	Instruments
Earth magnetospheric auroral imaging and geo-coronal emissions	UV camera
Sky large field imaging	Visible and UV cameras
Monitor of variability of selected cosmic X-ray sources (AGN's, cataclysmic variables, active binaries)	X-ray spectro-imager
Molecular line observation in selected bands (e.g. O_2 at 60 GHz)	Sub-mm receiver

Science payload selection

Following issue of the Announcement of Opportunity (AO) for SMART-1's scientific payload on 6 March 1998, ESA received fourteen proposals for instrument-based scientific investigations from European Principal Investigator (PI) teams, providing excellent coverage of the science and technology objectives described in the AO. In addition, three proposals and ideas for support science investigations involving no hardware were also received. The Agency established an independent Science Payload Peer Review Committee to assess the proposals and its recommendations have subsequently been endorsed by the ESA Advisory Groups – Solar System Working Group, Astronomy Working Group and Space Science Advisory Committee – and by the Science Programme Committee (SPC).

Evaluation criteria

The Science Payload Peer Review Committee rated the proposals according to the following criteria, as indicated in the AO:

– Science, technical, programmatic compatibility with SMART-1.
– Value for future Cornerstone missions (for example, mission to Mercury, deep-space astronomy missions).
– Support to technology demonstration and environment characterisation for the SEPP mission.
– Originality of science/technology with regard to current and planned missions.
– Relevance for mission scenario and proposed target.
– Demonstrated technological feasibility, reliability, readiness and development status of the proposed instrumentation.
– Competence and experience of the team in all relevant areas (science, management, space technology, proposed techniques, software development and technology, etc.).
– Adequacy of funding, manpower, management, schedule.
– Communication, public outreach and education aspects.

The Committee also set the following priorities for the mission:

• Rendezvous with Near-Earth Objects after Lunar Gravity Assist Flyby

The NEOs recommended, in order of interest, are Comet Asteroid Transition Objects and C-type asteroids, which are believed to represent the final phase in the evolution of dying comets and may constitute up to 40% of NEOs. Besides their interest for cometary and early Solar System understanding, they represent a class of objects that pose specific hazards to the Earth. They are indeed of low density and loosely bound, so that they could be split during close encounters with our planet, increasing the likelihood of later Earth impacts. It is important to characterise the internal and surface chemical and physical inhomogeneity of these objects. Other rocky or more rigid asteroids are, however, also of interest as they could represent the seeds from which the inner planets have accreted.

Target objects for rendezvous have been identified for a launch in 2001-2002 and a decision will be taken after an exhaustive search and optimisation effort. The lunar gravity assist might allow an enhanced payload and a shorter cruise phase, as well as an opportunity for a lunar polar science flyby. This would, for instance, allow one to pursue the remote exploration of permanently dark polar areas as possible ice reservoirs, and to validate some scientific results from the Lunar Prospector mission. High-resolution mapping of the crater-rim areas that are almost permanently sunlit is also of interest in the context of future lunar landings and outposts.

• Lunar Near-Polar Orbit

This option would allow an orbit with 1000 km perilune and 10 000 km apolune, thereby providing an additional lunar-science contribution to the flyby. The lowering of the orbit would cost significant fuel, but the high-resolution instruments would provide better data even from 1000 km perilune. The mission could then address several topical problems, such as: the accretional processes that led to the formation of the planets; the origin of the Earth-Moon system; the dichotomy between the far- and near-sides; the relatively long-term activity and the thermal and/or dynamic processes responsible for this volcanic and tectonic activity; and the external processes on the surface (impact craters, erosion and regolith formation, deposition of ice and volatiles).

The recommended science payload is compatible with both the asteroid and the lunar mission scenarios, the final choice depending on the choice of launcher and the resources available from the spacecraft. Both scenarios would provide extended periods for cruise-science astronomy observations.

The science model payload

A core science payload has been identified and is summarised in Table 6. It includes a multi-camera system with integrated electronics. The high-resolution GEMINI optics can provide 14 m definition at 1000 km or 0.4 m at 25 km (typical NEO rendezvous distance), fulfilling the

*Table 6. Science model payload**

Instrument	Mass [kg]	Comment	Cornerstone	Solar System	Astronomy
GEMINI : high res. optics	2.3	Wide FOV replaced by AMIE	X	XXX	X
SI : light optics	4.0	No scan unit, no electronics		XX	
Common SAGA electronics	2.4	SAGA control electronics and DPU	XX	XXX	X
AMIE WAC	0.8	Only 20° FOV, shared DPU	XX	XX	X
RSIS	0	Uses S/C X-Ka transponder	XXX	XX	FP**
SPEDE	0.5	Reduced boom to 0.5 m	XX	XX	
IXS light	3.4	20 cm focal length, 1-2 CCD	X	XX	XX
SMOG	3.3	Combined with S/C antenna	XX		XX

* The number of X's is an indication of each instrument's potential contribution to the future Cornerstones of Horizons 2000 and to the Solar System and Astronomical sciences.
**FP= Fundamental Physics

mission goals for both lunar and NEO science. It reuses a detector already being developed for Rosetta. The AMIE wide-field instrument uses a micro-camera system and includes a miniature data-processing unit that is of major interest for future missions. The SI visible near-infrared spectral imager will help in mapping mineral distributions on the targets. The electronics for these three channels will be integrated into a single package known as SAGA (SI-AMIE-GEMINI Assembly) sharing common subsystems. This not only has the advantage of reducing mass, but also paves the way for the integration and miniaturisation of the electronics required for future missions.

An Imaging X-ray Spectrometer (IXS) will provide X-ray images with energy discrimination, allowing mapping of the major elements on the target surfaces. It will also provide an opportunity for X-ray astronomy during the cruise phase, particularly temporal monitoring of X-ray sources in stellar clusters and Active Galactic Nuclei. It builds on the development effort for XMM's X-ray CCDs, and also serves to prepare for future Cornerstone missions.

A Radio Science Investigation (RSIS) using a planned technology X-Ka radio link permits the mass, moment of inertia and internal density distribution of the asteroid to be measured. It should also provide improved measurements of relativistic space-time curvature.

The lightweight Spacecraft Potential, Electron and Dust Experiment (SPEDE) will both characterise the plasma environment around the spacecraft and also provide useful scientific data.

The SMOG instrument is designed to detect and map galactic molecular oxygen with unprecedented sensitivity. It will also serve as a technological demonstration ahead of the FIRST and Planck scientific missions.

This model scientific payload is currently being verified in terms of spacecraft-resource and financial viability. Depending on the outcome, as well as on the technology payload allocation (see below), other potential scientific payload items have been earmarked, such as a Lyman-Alpha and UV mapper for measuring lunar or asteroid outgassing as well as astronomical observations, or additional micro-cameras on pointing micro-turrets.

The technology model payload

The SMART-1 payload will also include bus and instrument technology payloads, again selected during the design phase via an AO that was issued in April 1998. The technology proposals received by the 5 June deadline are described below. The electric propulsion system, however, was not part of the AO, and will be procured via a competitive Invitation to Tender (ITT).

Electric propulsion

The most important technology to be flown on SMART-1 is the Solar Electric Primary Propulsion (SEPP) system demonstration. Today, Europe already has a large inventory of electric thrusters either under development or already at the qualification stage, primarily for telecommunications spacecraft applications. Several of them are candidates for use as primary propulsion thrusters for deep-space missions the size of SMART-1, including the so-called stationary-plasma, radio-frequency-ionisation and electron-bombardment-ionisation types.

Stationary Plasma Thrusters are a family of electric propulsion engines belonging to the category of “Hall-effect Thrusters” (Fig. 5).

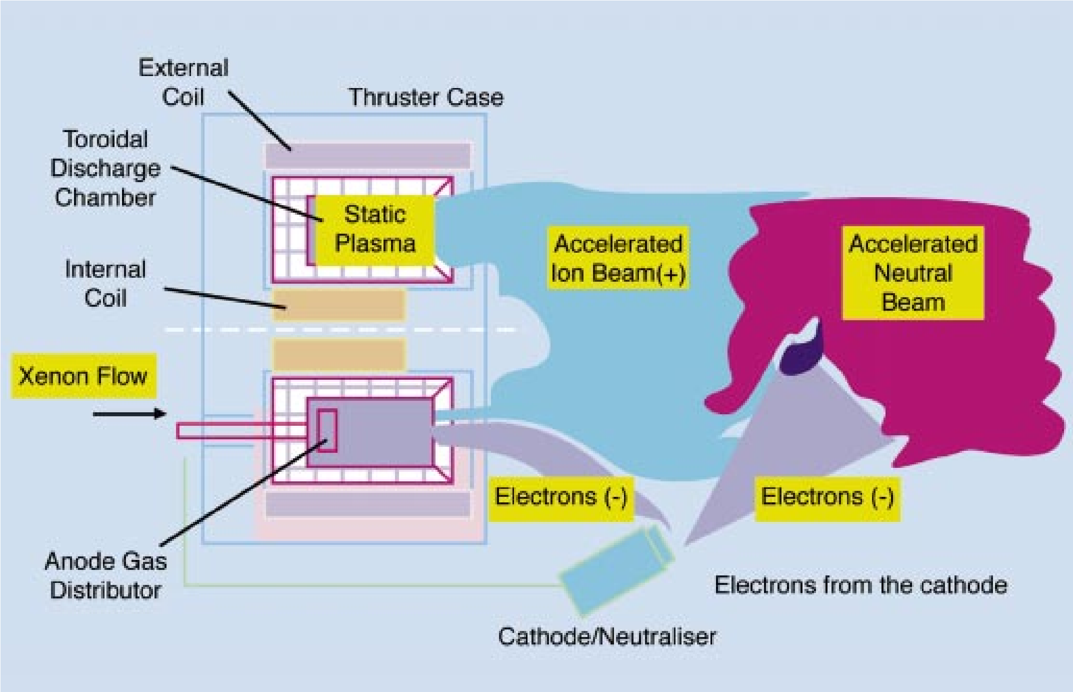

Figure 5. Schematic of a Stationary Plasma Thruster

Electrons from an external cathode enter a ceramic discharge chamber, attracted by an anode. On their way to the anode, the electrons encounter a radial magnetic field created between inner and outer coils, causing cyclotron motion around the magnetic field lines. Collisions between drifting electrons and xenon propellant create the plasma. The ions created are then accelerated by the negative potential existing near the exit of the chamber due to the Hall effect. The external cathode acts also as a neutraliser, injecting electrons into the thrust beam to maintain zero-charge equilibrium both in the beam and on the spacecraft. The PPS1350 has an exit diameter of 100 mm and provides a nominal thrust of 70 mN at 1640 s specific impulse (Isp) and 1350 W of nominal input power. This type of thruster, which can also work at reduced power, has already completed 7000 h of cyclic-operation qualification (corresponding to a total impulse of $2x10^6$ Ns).

Radio-frequency Ionisation Thrusters (Fig. 6) fall into the category of ion engines. The xenon propellant flows inside a ceramic discharge chamber through the extraction anode, which also serves as a gas distributor. The discharge chamber itself is surrounded by an induction coil connected to a radio-frequency (RF) generator. Free electrons within the xenon gas collect energy from the RF-induced electric field and ionise the neutral propellant atoms via inelastic collisions. The discharge is ignited by the injection of electrons from the neutraliser. Thrust is generated by the acceleration of ions

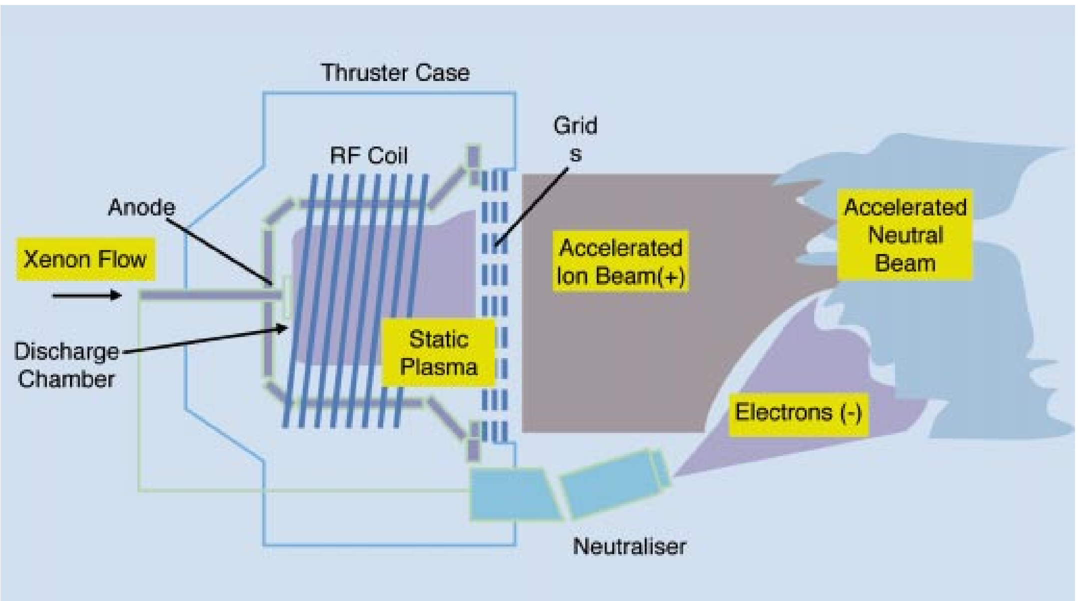

Figure 6. Schematic of a Radio-frequency Ionisation Thruster (RIT)

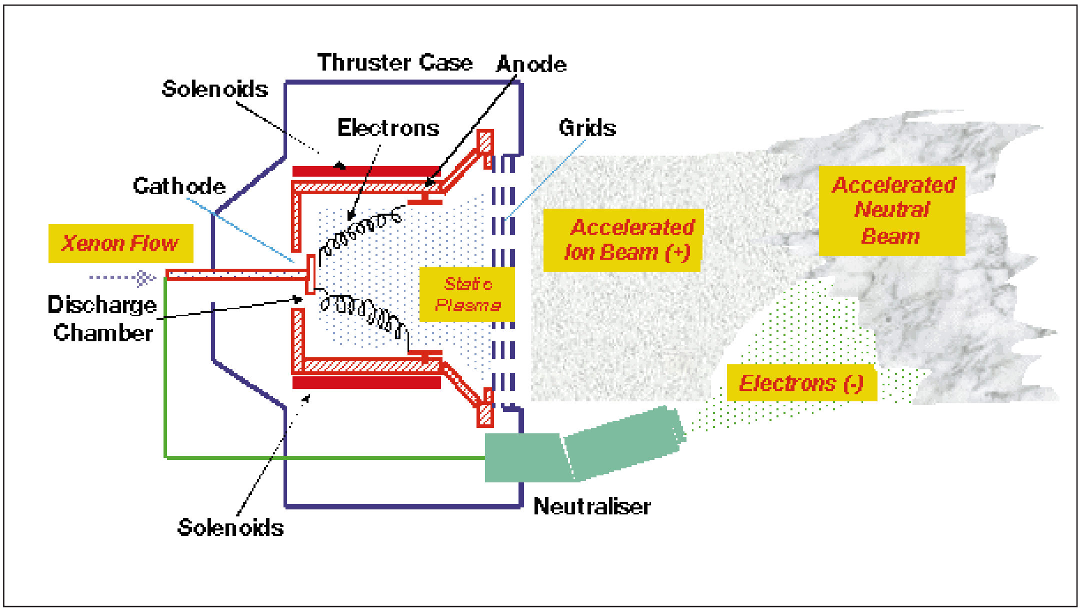

Figure 7. Schematic of an Electron Bombardment Ionisation Thruster

in the electrostatic field applied to an extraction system consisting of the extraction anode and three grids. The negative potential of these grids accelerates the positive ions out of the static plasma. A neutraliser injects electrons into the beam to maintain its zero-charge equilibrium and that of the spacecraft. The RIT-10 has an exit diameter of 100 mm and provides a maximum (modulatable) thrust of 23 mN at 3060 s Isp for an input power of 700 W. The thruster is presently being qualified for 15 000 h of cyclic operation (corresponding to a total impulse of $1x10^6$ N) at 15 mN.

Electron Bombardment Ionisation Thrusters (Fig. 7) also belong to the ion-engine category. In this case the xenon propellant flows inside a ceramic discharge chamber through a gas distributor. Free electrons produced by a cathode inside the chamber are attracted by an anode pole at the end of the chamber and flow along magnetic field lines created by a number of electromagnetic coils surrounding the chamber. Along this path, the electrons hit the propellant atoms and ionise them. Thrust is generated by the acceleration of the ions in the electrostatic field applied to an extraction system consisting of the extraction anode and three grids. The negative potential of the grids accelerates the positive ions out of the static plasma. A neutraliser injects electrons into the beam to maintain zero-charge equilibrium in the beam and on the spacecraft. The UK-10 has an exit diameter of 100 mm and the current version provides a maximum (modulatable) thrust of 23 mN at 3400 s Isp for an input power of 700 W.

Technology Announcement of Opportunity

Within the scope of the present AO for technology items, the following categories of Technology Experiments were considered:

– Key spacecraft technologies, which shall be prime constituents of the spacecraft or of one of its sub-systems. An example of such a technology item could be the Lithium-Carbon battery used as sole power source during eclipse and not backed-up by any conventional type of battery. An example of a complete spacecraft unit realised as a technology experiment could be a deep-space TT&C package, including an X/Ka-band transponder and antennas, if fully supporting all telemetry of the scientific data.

– Technology for science and technology experiments/instruments ancillary to or in support of the mission and of its prime objective (demonstration of SEPP). Examples of items falling into this category could be the InP MMIC front-end of the millimetre-wave radiometer, or a plasma-diagnostic package for characterising the solar electric propulsion environment.

– Technology experiments for spacecraft units, which are complements of on-board technology items to be operated as experiments, i.e. in parallel with a spacecraft unit/part realised with a conventional technology. One such example could be a novel type of gyro, operated in parallel with the nominal ACS device, to characterise and compare its in-flight performance, or a small technology experiment such as a miniature laser altimeter supporting autonomous planetary navigation.

Spacecraft procurement and management

Spacecraft design

The three-axis-stabilised spacecraft has been preliminarily designed for accommodation on Ariane-5 (Cyclade configuration) or within the Eurockot fairing. This requirement strongly constrains the geometric envelope to a cylinder 2.4 m in diameter and 1.0 m high, making the size of the solar panels a critical spacecraft design parameter. The maximum power available therefore ranges from 1300 W with GaAs/Ge cells, to 1500 W with GaInP/GaAs/Ge cells. The solar panels rotate to provide continuous tracking of the Sun during the mission, whilst also allowing rotation perpendicular to the solar vector to cover all thrust directions. Lithium-Carbon batteries will be used. A data-handling and attitude-control system based on that of the Odin spacecraft is foreseen. Star sensors will provide attitude information. Fibre-optic gyros or accelerometer packages are also foreseen for safe modes and rate damping. The actuators will be reaction wheels and mono-propellant hydrazine thrusters, in addition to the two-axis gimbals of the SEPP. The communication system is based on an S-band transponder supporting packetised command and telemetry. A X/X-Ka band transponder is also foreseen as a technology experiment, together with a TWT high-power amplifier. In the case of the NEO mission, the communication subsystem would be based on an S/S-X or X/X-band deep-space transponder and high-gain antennas. The platform's dry mass ranges between 250 and 290 kg, depending on the type of engine chosen and the degree of redundancy implemented. The amount of xenon fuel needed, and consequently the payload mass available, strongly depends on which mission is chosen. However, the maximum expected fuel load is 70 kg, and an upper limit of 20 kg for payload has been set.

Management approach

To achieve the mission objectives set, within the strict budgetary constraints and given the obligations of the ESA Scientific Programme, a project management plan has been devised based on the following guidelines:

- Full up-front ESA involvement in the design phase in an integrated team with the prime contractor, in order to ensure that the design fulfils the mission requirements and scientific and technology payload-selection criteria.
- Minimum ESA involvement during the development phase in non-critical activities, but with the delegation to the prime contractor allowing ESA full visibility.
- ESA involvement as the supplier of critical technologies.
- Full down-stream ESA involvement in the critical mission operations and in the assessment of the technological results.

This new management approach is aimed at maximising the use of the existing technical expertise, whilst reducing the management overhead, maintaining a high product standard and controlling the associated risks.

Acknowledgement

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