Marco Polo Assessment Study

Cosmic Vision 2015-2025

Executive Summary

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1 INTRODUCTION

In the frame of its Cosmic Vision 2015-2025 programme ESA has selected a number of mission concepts to be assessed by industrial studies. One of the concepts is the asteroid sample return mission named Marco Polo, having the primary scientific goal of returning a sample from a primitive near Earth asteroid to Earth. Global and local characterisation of the target asteroid is a further scientific requirement of the mission, with focus on the context determination of the sampling site. Marco Polo has been proposed as an M-class joint mission by the European Space Agency within the Cosmic Vision programme and by the Japanese Space Exploration Agency.

Various options are being contemplated in the frame of this collaboration. This executive summary presents the results of the Marco Polo study performed under ESA contract by the industrial team led by OHB-System AG, for an ESA-defined scenario. Possible collaboration schemes associated with this scenario are not addressed here. The results presented address the mission and system design of the ESA space element of the Marco Polo mission including the analysis of critical technologies required for this ambitious mission.

The industrial team was led by OHB-System AG and included the following partners:

- GMV S.A. – focussing on mission analysis and GNC technologies
- Sener S.A. – focussing on landing and sample acquisition technologies
- Aero Sekur S.p.A. – focussing on high speed re-entry technologies
- QinetiQ Ltd – providing consultancy in the area of electric propulsion

2 MISSION ARCHITECTURE TRADE-OFF

The first part of the performed study was dedicated to a comprehensive trade-off process on mission and system architecture level, including the critical technologies. High level trade-offs were performed with respect to the following major mission aspects.

Target Asteroid

Four primary targets, being 1989 UQ, 1999 JU3, 2001 SG286 and 2001 SK162 have been considered. In addition, some further target asteroids have been assessed. Finally the asteroid 1999 JU3 has been selected as target of the Marco Polo mission. This choice was mainly driven by its easy accessibility with spacecraft purely based on chemical propulsion, favourable launch dates and mission durations in combination with asteroid stay times comfortably allowing both the sample acquisition and remote science of the target.

Propulsion Technology & Interplanetary Transfer

Mission scenarios based on the usage of chemical propulsion and electric propulsion have been studied. Combinations of both propulsion technologies were also investigated. Different launch opportunities between 2017 and 2020 have been analysed. In combination with the number of possible asteroid targets and the existing propulsion technology options, a high number of interplanetary trajectories have been optimised.

Finally, in combination with the choice of the target asteroid, chemical propulsion has been selected for the Marco Polo mission, with the baseline launch in November 2018 and a backup launch opportunity one year later. The resulting mission duration is six years, with Earth return in December 2024. Details of the interplanetary transfer are given below.

Landing & Sampling Strategy

The landing and sampling approach has been analyses in detail at both technology and system level. In total, five options have been studied.

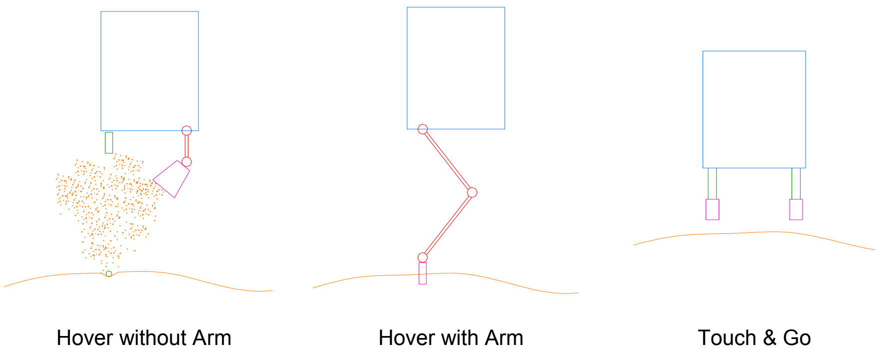

Figure 1: Hovering and Touch & Go

In case of *hovering*, the sampling spacecraft keeps its position over the asteroid surface while the sample is taken. There is no need for a landing system (landing legs) but the option is the most demanding one from the GNC point of view.

Two suboptions exist for the acquisition of the sample. First, a projectile type tool can be used to generate dust which is then collected by suitable device accommodated on the sampling spacecraft (Figure 1, left). The sample mass is very limited in this case. Second, a sampling arm can be used (Figure 1, middle).

Another possible option is to combine landing and sampling in that sense that landing legs are equipped with sampling devices and the impact energy of the touch-down is used to take the sample. This approach is commonly referred to as touch & go. It is the simplest option for GNC, but requires the most complex landing and sampling system (Figure 1, right).

The full landing option is a compromise between hovering and touch & go in that sense, that a landing is performed but the landing system is mechanically separated from the sampling system (Figure 2). The challenge of this option is to ensure that the sampling spacecraft stays stabilised on the asteroid surface during the sampling operations, as the sampling forces will be higher than the gravity forces of a low gravity body.

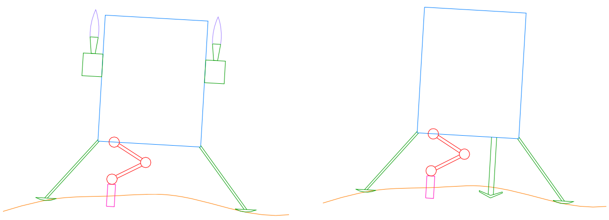

Figure 2: Full Landing Option

Also in this case, two suboptions exist. First, surface time can be limited to some tens of minutes, eliminating the need for an anchoring system. During this time, the sampling spacecraft is kept on the asteroid surface by firing RCS thrusters oriented against the asteroid (left part of Figure 2). Second, an anchoring device can be implemented (right part of Figure 2).

The anchoring option is, however, characterised by a high risk due to unknown surface characteristics of the target asteroid.

Finally, full landing with short stay on the surface and use of RCS thrusters for sampling spacecraft stabilisation has been baselined for Marco Polo.

System Configuration

A systematic trade-off process was conducted for the definition of the Marco Polo spacecraft configuration. Resulting from the numerous different functions to be covered by the space segment of a sample return mission (interplanetary flight, asteroid orbiting, landing and sampling, return to Earth and re-entry), a high number of possibilities exist and was explored.

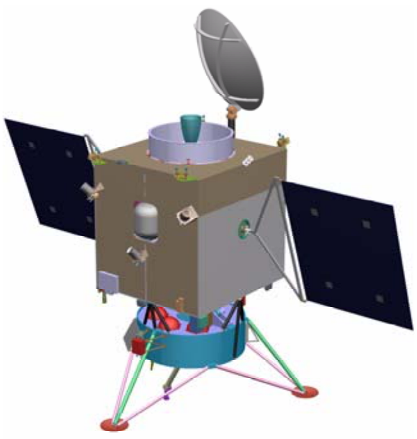

Figure 3: Configuration Option 1

Obviously, Earth re-entry will be performed by a dedicated capsule, so that one of the space segment modules is defined by the capsule. To perform the remaining functions of the mission, two module configurations as well as a single module configuration were studied on conceptual level.

If implementing the two modules configuration, the first option is to separate the interplanetary transfer (outbound and inbound) and asteroid orbiting from the landing and sampling operations. This leads to the composite concept as illustrated in Figure 3. The upper part of the composite is the orbiter module performing the interplanetary transfers and asteroid remote sensing operations.

A second option is to separate the interplanetary outbound transfer and orbiting operations from the landing/sampling as well as interplanetary inbound transfer function. A space segment concept for this case is presented in the left part of Figure 4. The advantage of this option is the fact, that remote sensing science at the asteroid can be continued while the sample is already on the way back to Earth.

A third option is to separate the interplanetary outbound transfer and sampling/landing from the orbiting function and Earth return. An exemplary design concept for this case is shown in the right part of Figure 4.

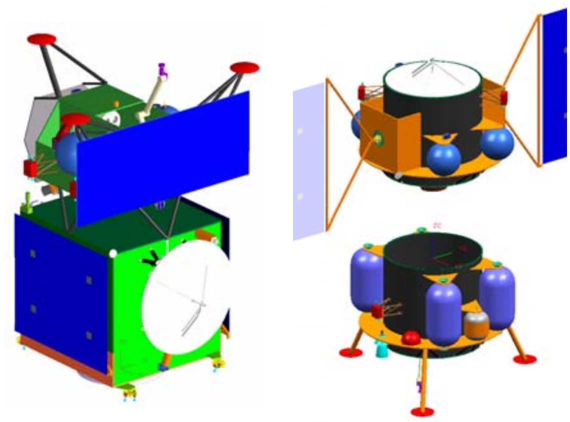

Figure 4: Configuration Option 2 and 3

In all the above cases, the re-entry capsule is accommodated between the two modules.

Finally, a single module configuration has been selected (sampling spacecraft and re-entry capsule), driven by programmatic mission aspects. This baseline system configuration is described in more detail below.

3 MISSION ANALYSIS

The selected Marco Polo target asteroid 1999 JU3 is an Apollo asteroid with an orbit that grazes both the orbit of Earth and that of Mars, and which is inclined some 5° with respect to the ecliptic plane. The perihelion is located close to the descending node, so that a launch or a final Earth swingby close to it (around December) can be used to provide the needed aphelion and inclination change.

The relative phasing is favourable for an Earth gravity assist in December 2020, and the outbound transfers selected as the nominal and backup trajectories for Marco Polo exploit this phasing opportunity by performing a final Earth swingby at that date. The backup trajectory is launched one year before (Nov-2019), while the nominal trajectory is launched two years before (Nov-2018) and includes an intermediate Earth swingby. From the final Earth gravity assist onwards, the two missions are identical. In both cases, the spacecraft arrives at the asteroid in Feb-2022 and performs an arrival manoeuvre slightly higher than 100 m/s to cancel the relative velocity (see Figure 5).

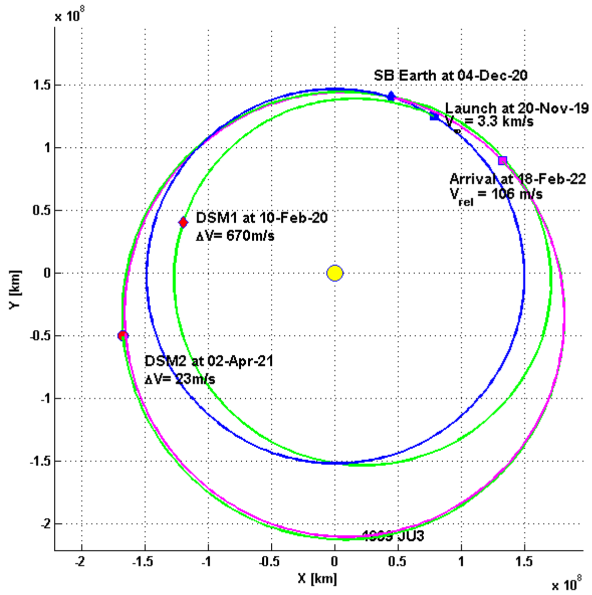

Figure 5: Outbound Transfer in 2019

The initial Earth-Earth arcs of the nominal and back-up trajectories give considerable flexibility to the mission design. They can be used for leveraging the spacecraft heliocentric velocity through a technique known as DV-EGA (ΔV Earth Gravity Assist), which consists in inserting a number of Deep Space Manoeuvres (DSM) in order to reduce the launch escape velocity.

Depending on the performance characteristics of the launcher and the specific impulse of the spacecraft propulsion system, this technique can result in higher final spacecraft masses at the asteroid, at the cost of a higher ΔV budget to be provided by the spacecraft (larger propellant tanks). A trade-off was performed at system level and the ΔV budget for the outbound interplanetary transfer has been set to 900 m/s.

Interplanetary Trajectory Characteristics	
Nominal launch date	Nov-2018
Backup launch date	Nov-2019
Escape velocity [km/s]	2.96 – 3.05
Escape declination [deg]	0.0
First Earth swingby date (nominal)	Nov/Dec-2019
ΔV before final Earth swingby [km/s]	0.66 – 0.69
Final Earth swingby date	04/12/2020
Arrival date	Jan – Feb 2022
Arrival ΔV [km/s]	0.1-0.13
Manoeuvre budget [km/s]	0.9
Asteroid stay time [months]	18
Asteroid departure date	30-Jul-2023
Asteroid departure ΔV [km/s]	0.500
Earth arrival date	04-Dec-2024
Hyperbolic arrival velocity [km/s]	4.991
Total mission duration [years]	5.03

Table 1: Interplanetary Trajectory Characteristics

The return trajectory makes use of the same favourable geometry, i.e. arriving at the Earth close to the perihelion of the asteroid, at the second next favourable phasing opportunity, which corresponds to an Earth arrival in December 2024. The relative arrival velocity at the Earth is slightly below 5 km/s, which leads to atmospheric entry velocities ranging between 11.75 km/s and 12.4 km/s, depending on the latitude of the entry point. Table 1 summarizes the characteristics of the Marco Polo interplanetary trajectory for the backup launch in 2019.

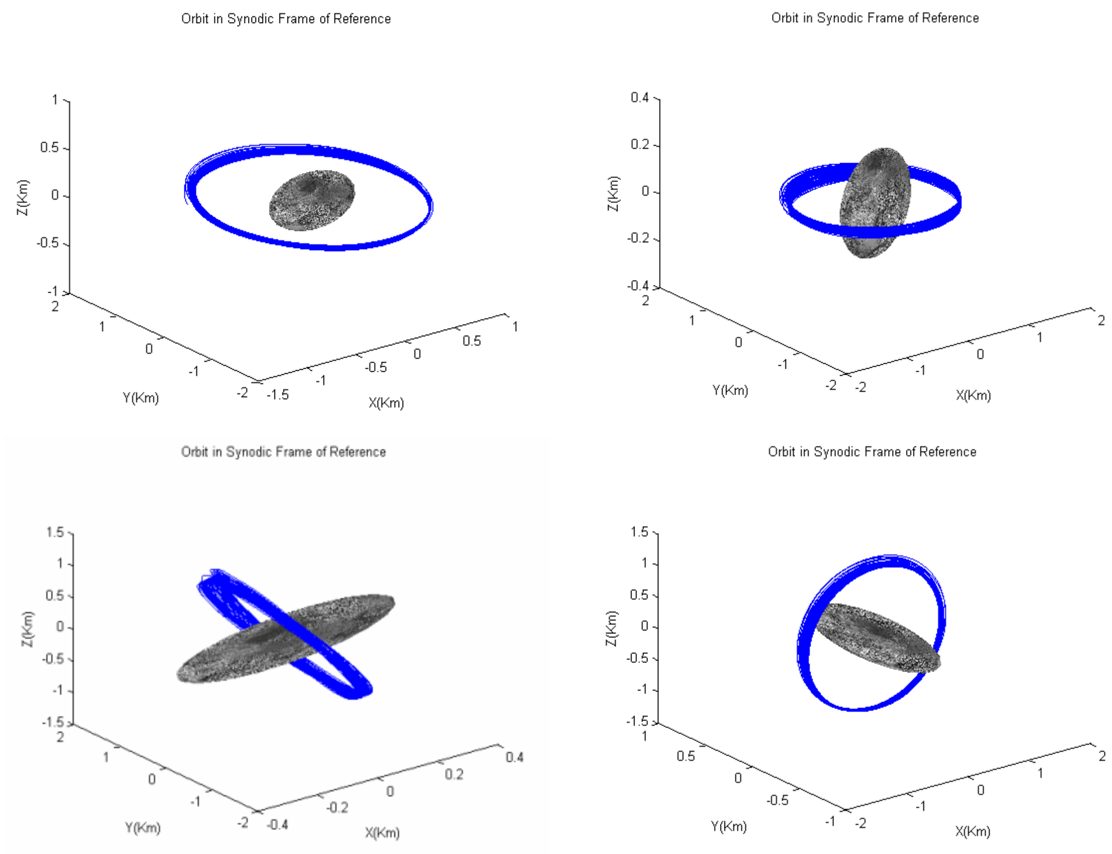

Figure 6: Examples of Closed Controlled Orbits

At the beginning of the scientific phase, there will be still much uncertainty concerning the main physical properties of the asteroid (size, shape, gravitational field, rotational state). Therefore, the spacecraft will first remain at higher (safer) distances, and then it will get closer progressively as the models of the dynamics are refined. The different orbital strategies baselined for the scientific phase are:

- Far and close formation flying – These are orbital strategies where the spacecraft maintains its relative position with respect to the asteroid, either in a near-inertial frame (synodic), or in a local frame rotating with the asteroid (*body hovering*).
- Self-stabilizing terminator orbits (SSTO) – For small asteroids, the combined effects of the asteroid gravitational attraction and the solar radiation pressure give emergence to a special type of orbits: SSTO (or dawn-dusk). These orbits are free from eclipses and they do not require orbit maintenance.
- Close controlled orbits – Close orbits (see Figure 6) have the main benefit of the increased resolution of the scientific observations, but the closer to the asteroid the lower the limit of stability for SSTO. They are not intrinsically stable and must be controlled.
- Descent and landing trajectories – The low gravitation allows for planning radial descent trajectories (no eclipses, simple attitude control), defined by several control points where the spacecraft will hover to wait for ground confirmation for continuing the descent.

4 SAMPLING SPACECRAFT DESIGN

The Marco Polo space segment consists of the two spacecraft modules:

- Sampling spacecraft – performing all mission functions except for Earth re-entry
- Earth re-entry capsule – performing atmospheric Earth re-entry and landing as last part of the mission

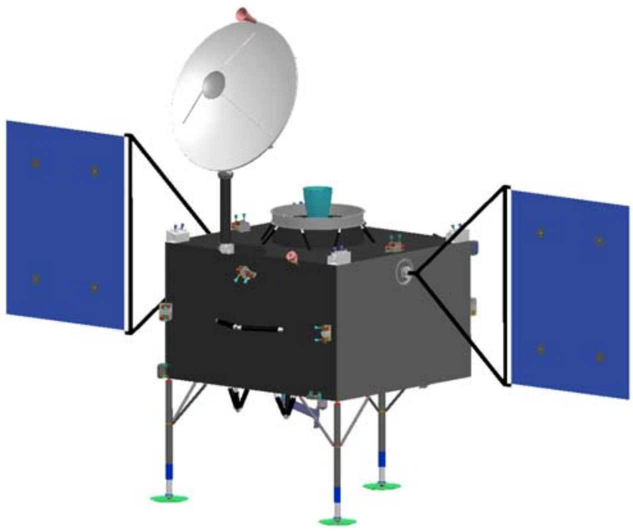

Figure 7: Sampling Spacecraft (1)

The mechanical layout of the sampling spacecraft is based on a simple structural concept, using a central tube as the primary load carrying structure and six main outer panels forming the spacecraft box with dimensions of 2350 mm x 2100 mm x 1400 mm. The sampling spacecraft dry mass is around 725 kg, including science payload instruments and design margins.

Two deployable, rotating solar generator wings of 3.2 m² area each are used as primary energy source, avoiding power limitations for the system. As result, high RF power can be implemented, enabling high data rates for the transmission of the science data to Earth, and therefore increasing the overall science return from the mission. Rotating solar generators also give high spacecraft pointing flexibility during science observations at the target asteroid as well as enable the spacecraft power supply from the solar generator during the first phase of descent and landing. An on-board battery covers the remaining part of the descent, the whole surface operations phase and the following ascent, up to reaching the sun pointing attitude.

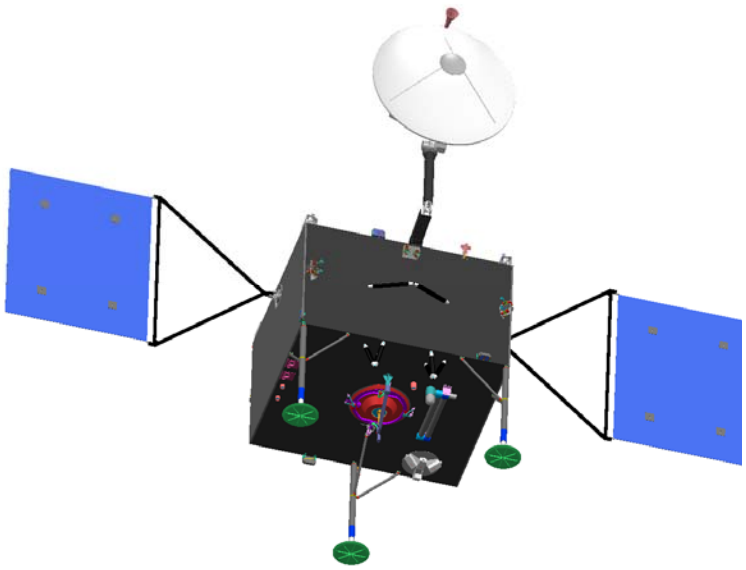

Figure 8: Sampling Spacecraft (2)

The sampling spacecraft is equipped with two separate propulsion systems. A standard bi-propulsion system is used for the interplanetary transfer and propulsive manoeuvres above ~1 km altitude over the asteroid surface. Below this limit, the mono-propulsion system based on hydrazine is operated to perform all propulsive manoeuvres. In this way, asteroid surface contamination by bi-propellants is avoided. The bi-propulsion system includes a main engine to benefit from its high efficiency. However, if mass margins allow in future, the engine could be removed from the system without any redesign effort.

Aiming at maximising the data rate for science data transmission to Earth, the sampling spacecraft is equipped with a 1.6 m HGA. A compact design of the main spacecraft enables the accommodation of this HGA under the Soyuz/Fregat launcher faring. The antenna set is completed by an MGA and two LGAs for data transmission during landing and for contingency cases. X-band with 120 W nominal RF power is baselined.

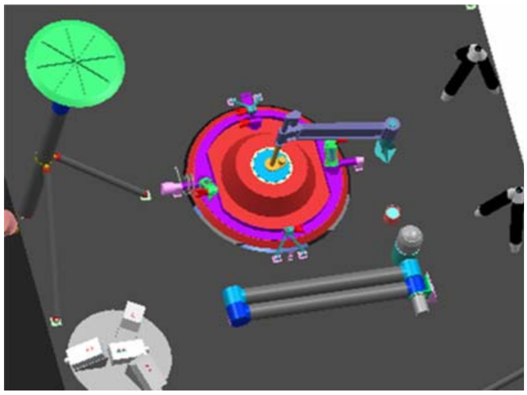

Figure 9: Sampling System

The AOCS/GNC subsystem is based on star trackers, IMUs, coarse sun sensors and reaction wheels for the standard operations during interplanetary transfer. For asteroid observation phases, and in particular for descent and landing, the system is completed by four navigation cameras, two laser altimeters and a set of short range radar altimeters in a configuration to provide relative altitude and attitude information during the final part of the descent.

The Earth Re-entry Capsule (ERC) is accommodated on the surface deck (the one closest to asteroid surface at landing) of the sampling spacecraft. As consequence of the resulting short transport way for the asteroid sample from surface to the sample container inside ERC, the sampling and transfer system can be reduced in complexity. It consists of only two major elements, the extendable sampling arm and a second arm used to close the sample container and the ERC.

Another advantage of the accommodation of the ERC on the spacecraft surface deck is the fact that the opposite panel can be easily used as launch deck. The spacecraft composite is launched in the configuration with landing legs oriented upwards, removing the need for a dedicated adapter structure between the spacecraft and the launcher adapter as well as for landing legs deployment mechanisms.

The Marco Polo science payload suite consists of the following seven instruments:

- Wide Angle Camera (WAC)
- Narrow Angle Camera (NAC)
- Laser Altimeter (LA)
- VisNIR Spectrometer (VisNIR)
- MidIR Spectrometer (MidIR)
- Neutral Particle Analyser (NPA)
- Close-Up Camera (CUC)

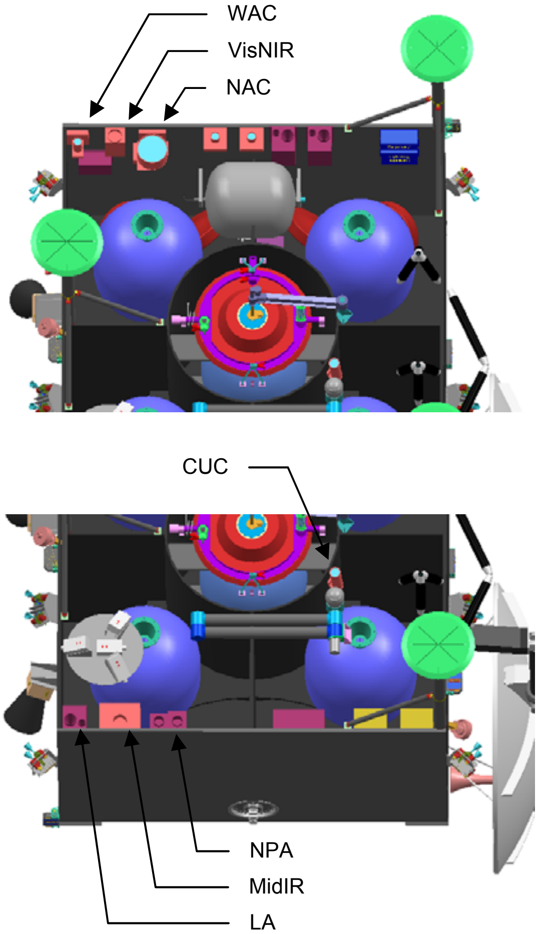

Figure 10: Science Payload

Another mandatory science experiment is the Radio Science Experiment (RSE). It is performed using the spacecraft communication subsystem so that no additional instrumentation is required. All science instruments are accommodated on the surface deck ensuring their field of view direction being in line with the field of view of the GNC sensors. This allows the science observations both from close asteroid orbits as well as in formation flying configuration (with spacecraft kept between asteroid and Sun for optimisation of the science data return). Figure 10 shows the accommodation of the Marco Polo science payloads on the sampling spacecraft. The CUC is accommodated close to the sampling arm to allow imaging of the sampling site before and after sample acquisition.

5 RE-ENTRY CAPSULE DESIGN

The Marco Polo Earth Re-entry Capsule (ERC) has the main function to safely return the asteroid sample to Earth. The main requirement that the capsule must satisfy is to guarantee the integrity of the sample container during the hot phase of the re-entry into the Earth atmosphere, the deceleration loads and the landing phase. The main technical drivers result from the interplanetary Earth return scenario and the scientific requirements of the asteroid sample to be brought to Earth:

- Atmospheric entry velocity – a value of 11.9 km/s results from the interplanetary transfer back to Earth for the launch windows baselined
- Heat flux and heat load – the maximum heat flux during re-entry was limited to 15 MW/m² and the total heat load to 250 MJ/m² (including margins)
- Sample container size – the sample container is sized to fulfil the science requirements with respect to the sample mass brought to Earth
- Sample temperature limits – the science requirements limit the maximum temperature of the sample to 40°C, with up to 80°C allowed for less than one minute

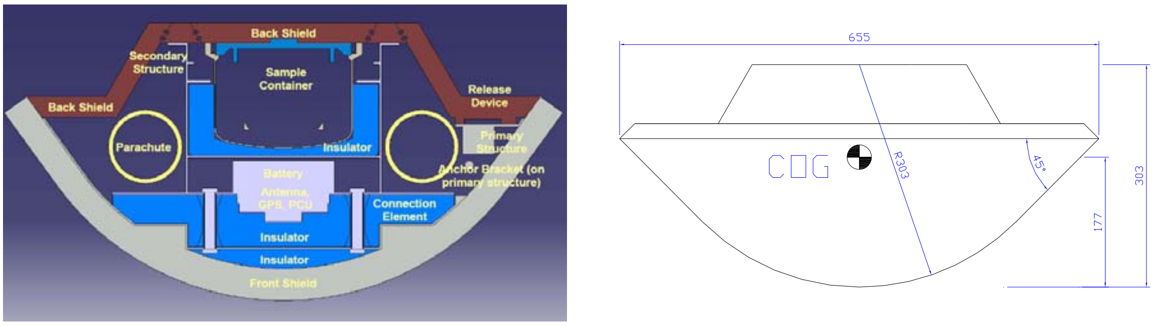

Figure 11: ERC Configuration

The obtained Marco Polo ERC configuration is presented in Figure 11, including the capsule main dimensions. The design is based on a half-cone angle of 45°, parachute based landing to limit impact shock loads and a non-separable front shield to decrease system complexity.

To guarantee the fulfilment of the thermal requirements and to ensure the integrity of the sample container during the hot phase of the re-entry, as well as to limit the ERC mass budget, a European ablative material has been considered as TPS material for the front shield, implementing a new concept of lightweight carbon-impregnated ablator with high efficiency and very low density when compared to traditional carbon based ablators.

6 SAMPLING AND TRANSFER SYSTEM

An extendable sampling arm has been baselined for Marco Polo following an extensive trade-off study. A one degree of freedom arm and a rotating corer with claws has been selected as primary sampling mechanism. A sticky pad backup sampling method was also included in the mechanism design of the Sample Acquisition and Transfer System (SAT).

The SAT system has to perform a number of independent functions. Consequently, the system has been divided in independent mechanisms performing these functions. The decomposition leads to easier and more flexible manufacturing and qualification campaigns. These functions and mechanisms are the extension of the corer, sampling, transferring and closure of the sample container.

Figure 12: Sampling Arm and Closing Arm

The extendable arm is based on two segments of the same length (left part of Figure 12). One extreme can rotate by the action of an electric motor. The central hinge (elbow) induces a rotation to the second segment that is synchronized with the driver in such a way that the rotation of the elbow is double than the driver but opposite direction. A similar synchronization device is used to move the head of the arm at the same angle to the driver. The motion is controlled by a unique driver located close to the spacecraft. This has the advantage that the driver is easy to thermally isolate, minimises harness and mass of the arm and optimises redundancy.

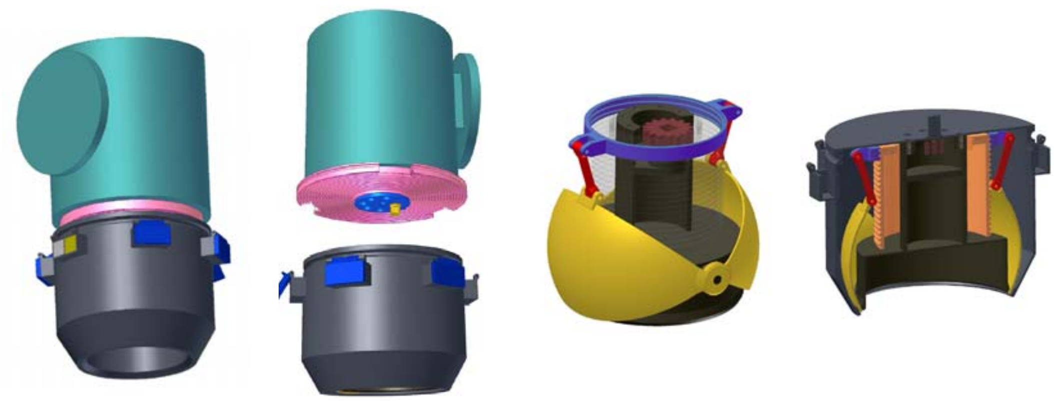

Figure 13: Corer with Sticky Pad

The corer is the key component of the sample acquisition mechanism, shown in Figure 13. It is a metallic cylindrical recipe used to penetrate the soil and acquire the sample. It has permanently attached witness plates to measure the contamination of the environment in the vicinity of the sampling site. A sticky pad (in pink colour in Figure 13) is located between the corer and the driver case. Its purpose is to be used as backup sampling device to acquire some reduced amount of sample in the case of failure of the corer or due to unexpected characteristics of the soil not compatible with the sampling philosophy. If the corer fails, the corer is released and the sticky surface is pressed onto the soil surface. After that, it is inserted into the ERC.

The closing arm is the second arm of the sample acquisition and transfer system (right part of Figure 12). It is used to transfer the corer into the ERC and to close the sample container via the ERC cover.

The SAT operations are mainly performed during the landing phase. It is foreseen to have continuous operations over 20 minutes of surface stay time.

7 LANDING SYSTEM

Three identical landing legs provide landing shock attenuation as well as spacecraft stability during the surface operations. The legs are designed to support up to three landings, by implementing crushable materials in the main struts. After touch-down, the spacecraft is kept on the asteroid surface by firing RCS thrusters providing acceleration towards the asteroid.

Figure 14: Attenuation Device Concept

The legs are identical to simplify design, manufacturing and verification process. The dimensions of the spacecraft allow good stability without the need of diverge arrangement of the legs for larger footprint surface. Stowing for launch is not required. The legs are designed to withstand the launch loads in fixed deployed configuration.

The low number landing attempts (i.e. three) and the low energy to be dissipated during landings allows the use of crunchable honeycomb cartridges (Figure 14) for vertical attenuation, benefiting from their advantages like excellent energy absorption performances due to the plastic behaviour of the honeycomb, very good force prediction that has low sensitivity to thermal conditions and contamination, low mass and high technology readiness level. The landing legs are supported at hard points of the sampling spacecraft main structure.

8 ERC RELEASE SYSTEM

System level considerations led to the request to accommodate the re-entry capsule on the surface deck of the sampling spacecraft. The major advantage of this concept is the short transfer way for the asteroid sample from surface to the sample container inside the ERC. The complexity of the SAT system is substantially reduced in this way. A further important advantage is the availability of the central part of the sampling spacecraft for accommodation of propellant tanks as well as flexibility to implement a main engine to increase the propellant efficiency of the spacecraft.

Also, sampling spacecraft deck opposite to the surface deck can be used as launch deck, allowing the use of a standard launch adapter and eliminating the need for dedicated launch support structures or deployment mechanisms for the landing legs.

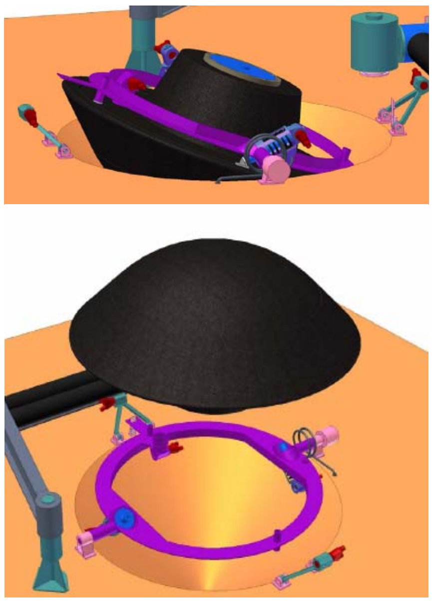

Figure 15: ERC Release Concept

The design solution implementing this requirement is illustrated in Figure 15. The ERC is accommodated on the surface deck of the sampling spacecraft via a supporting structural ring (in pink colour). For ERC release for re-entry, the ring with the attached ERC is rotated by 180°, allowing capsule ejection along the spacecraft velocity vector.

9 GNC SYSTEM

The mission requirements, the environment uncertainties and the programmatic constraints pose very demanding challenges on the GNC system design. In particular, the required landing accuracy is 3.5 m (3σ), the lowest ever for a spacecraft landing on a celestial body.

All mission phases during the proximity operations were analysed and a baseline GNC system has been designed for each mode as a compromise between cost and risk.

The GNC sensor suite consists of:

- 2 redundant laser altimeters used at distances larger than ~100 m (low gate where the fully autonomous descent and landing is initiated)
- 1 miniature radar altimeter (MRA) for distances below the low gate
- 3 MRAs surrounding the central one and tilted wrt nadir to measure the surface-relative attitude (and provide redundant altitude measurements), activated at lower altitude than the nadir-pointing one
- 2 redundant, nadir-pointing WACs for absolute and relative navigation
- 2 redundant, top-mounted, tilted WACs for contingency and surface operations
- 2 redundant star trackers for attitude determination and vision-based navigation during the asteroid approach phase
- 2 redundant inertial measurement units

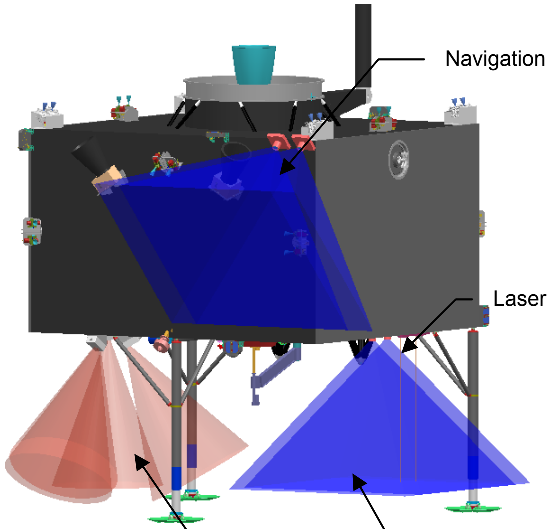

Figure 16: GNC Sensor Suite

Nominal Landing Performances			
Magnitude	Mean	Mean+3σ	Requirement
Landing accuracy (m)	1.15	2.89	3.5 (3σ)
Horizontal velocity (cm/s)	1.07	2.72	< 5
Vertical velocity (cm/s)	6.78	7.77	< 30
ΔV (m/s)	1.22	1.46	-

Table 2: Nominal Landing Performances

The actuators consist of a set of thrusters providing force and torque in any direction with no need of rotations (not to jeopardize the feature tracking carried out by the image processing function), and 4 reaction wheels to compensate attitude disturbances.

The descent and landing phase was considered the most critical one and a thorough analysis including extensive simulation campaigns was performed. The weak dynamics allows the definition of the spacecraft path by a set of waypoints with associated times. This strategy provides high flexibility in the definition of the descent profile in order to adapt to any asteroid shape and operational constraints.

For the descent and landing part that is above the low gate, the ground involvement is intended for navigation aiding and the descent strategy is based on small impulses that take the spacecraft from waypoint to waypoint. Intermediate corrective manoeuvres compensate navigation and execution errors. The close descent (below the low gate) must be carried out fully autonomously due to communication delays. The descent strategy consists of thrust arcs including a non-negligible portion of the flight time between the waypoints and allowing a continuous control of the descent.

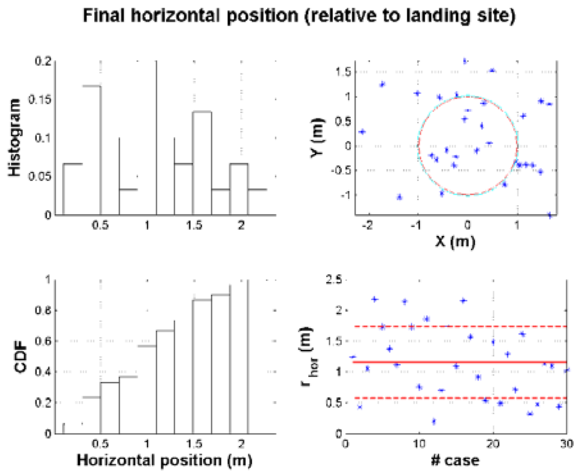

Figure 17: Nominal Landing Dispersion

The GNC system has been tested in closed-loop in a mission performance simulator with a benchmark scenario envisaged to prove the capabilities of the GNC in non benign conditions. Monte Carlo simulations were carried out to test the robustness of the GNC and for sensitivity analysis of mission and system parameters. The performances in nominal conditions are within the mission requirements as shown in Table 2. Landing dispersion is presented in Figure 17.

10 CONCLUSIONS

The industrial assessment study of the Marco Polo mission concept started with a comprehensive trade-off on mission and system level. As result, the mission architecture was defined, including the selection of 1999 JU3 as target asteroid, the definition of the baseline landing and sampling approach as well as the selection of the baseline system configuration.

The following analysis and design phase of the study led to a detailed design of the mission and the space segment, including the sampling spacecraft and the re-entry capsule. Critical technologies have been deeply analysed and a baseline solution taking into account the programmatic mission constraints has been defined for each technology.

The study results clearly show the feasibility of the Marco Polo mission concept within ESA's Cosmic Vision 2015-2025 context. Significant mass margins exist on mission level and the required technology readiness level can be reached for all critical technologies required for the mission.

At the same time, the proposed space segment design is characterised by high performance characteristics (multiple landings, high data rates, flexibility with respect to orbit selection) as well as high design flexibility. The latter point allows reacting to possible future changes of the mission requirements. This is considered a significant benefit for a mission concept today studied at assessment level.