Title: SCORE: Observation and Exploration of a Long Period Comet using Micro-Satellites
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Mission Objectives (where and why?)

Comets have always been interesting celestial objects during the history of mankind, as they are the most primitive objects in the Solar System, being able to provide the scientific community with information about its origin and evolution. Long Period Comets (LPCs) are characterized by an orbital period greater than 200 years. Differently from Short Period Comets, LPCs have not been contaminated by the various passages within the inner Solar System. SCORE (Scientific mission for Comet Observation, Research and Exploration) has been designed with the aim of intercepting an LPC in a fly-by scenario, gathering scientific information about these unexplored objects. Thanks to the development of space technologies applied to microsatellites, this goal can be achieved by using spacecrafts with reduced weight and size, while maintaining a high scientific return of the mission. If no LPC are identified in a useful time window, the spacecraft will be redirected to collect information about specific back-up targets. This allows to guarantee a significant scientific return from the mission. An extensive Market Analysis has been performed with the aim to identify the main interests of the scientific community toward comets, determining SCORE's Mission Objectives. The aim of the fly-by is to collect data about the comet's structure and morphology. The physical, chemical and dynamic properties of the celestial body shall be measured, estimating the magnetic field, the rotational speed, the gases and dusts in the coma and the nucleus composition. The comparison between scientific data collected by the SCORE mission and the data coming from similar missions such as Giotto, Comet Interceptor and Rosetta, will allow a deeper knowledge about the cometary environment, the genesis and the evolution of these celestial bodies, as long as important information about the development of the Solar System and the origin of the life on Earth. Technologies developed and improved in this mission are of great value for future deep space exploration missions with micro-satellite, started with MarCO mission[1]. In this way, doors are opened to low-cost missions towards other celestial objects, such as Mars, also supporting the human deep space exploration expected in the next decades.

Concept of Operations including orbital design

The spacecraft is deployed in the cis-lunar space, e.g. by the Lunar Gateway, during 2029. Until the identification of a potential target, the spacecraft waits on a parking *L2 Southern Near Rectilinear Halo Orbit (NRHO)*, whose perilune radius is 16000 km. This orbit is characterized by a very high stability, needing a *very low delta-V* for station-keeping maneuvers. Eclipses are rare and short, lasting about 160 min and occurring every 1-2 lunar synodic periods. In Figure 1 an example of the parking NRHO is shown. The region of study in which the mission is designed to perform the fly-by is a *Sun-centered annulus* of radius in between 1 and 1.58 AU. A *probabilistic analysis* has been carried out, starting from the orbital parameters of already known comets – properly selected from a NASA online database[2]. The *maximum time between two discoveries* of 2.1 years has been obtained. During this time, the probability of discovering a suitable LPC is higher than the 99%.

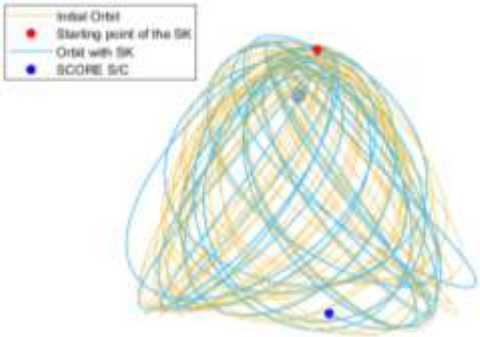

Figure 1: Example of the parking NRHO.

If a suitable target is detected, the satellite shall leave the parking NRHO. However, if in this period no targets are discovered a transfer towards a back-up target shall be planned. In Figure 2 the different phases of the mission are represented. The lifetime of the mission is expected to be of 6 years. The departure maneuver from the NRHO starts when the Moon is *in between the Sun and the Earth*. Since the orbit has a *high perilune radius*, the spacecraft will reach very high altitudes, almost comparable to the Moon's *Sphere of Influence* (SOI). Therefore, the departure delta-V for leaving the parking orbit will be low. Then, the spacecraft shall *depart* from the Earth's SOI. The satellite performs an impulsive maneuver, entering a transfer ellipse. The time required to the spacecraft to reach the boundary of the Earth's SOI is about 30.5 days. Then, an impulse of negligible magnitude concludes the departure phase and the interplanetary transfer towards the comet begins. Once every Moon's synodic period (29.5 days) these departure maneuvers can be made, using chemical propulsion. Electric propulsion is needed to perform the *interplanetary transfer* to the comet. Two transfer strategies are proposed. The first with a single finite burn, reaching a maximum distance of 1.39 AU from the Sun (Figure 3, left). The second with a double finite burn, allowing the satellite to reach distances up to 1.58 AU (Figure 3, right). The *flyby* between the comet and the spacecraft occurs in the ecliptic plane and the main role is played by the relative speed between the spacecraft and the comet. The maximum relative velocity has been obtained analyzing comets of interest and is estimated to be slightly higher than 70 km/s. The fly-by duration is of a few minutes. Once the scientific data has been stored, the satellite must *transmit* them to the selected ground stations (New Norcia and Malargue[3]). To have a high data rate, the satellite shall not be too far from the Earth. If the data rate is low, additional maneuvers may be performed to bring the satellite closer to the

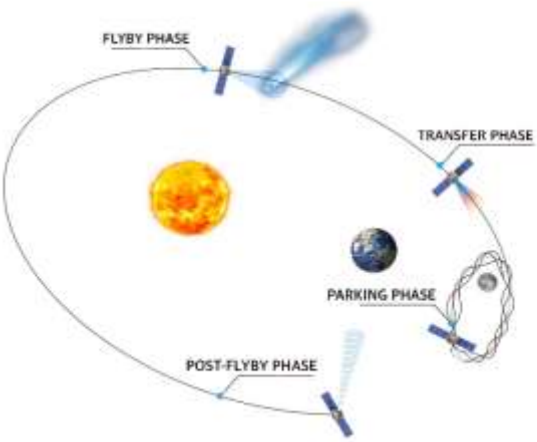

Figure 2: Mission operations phases.

Earth. Orbital design has been performed through the *GMAT software.*

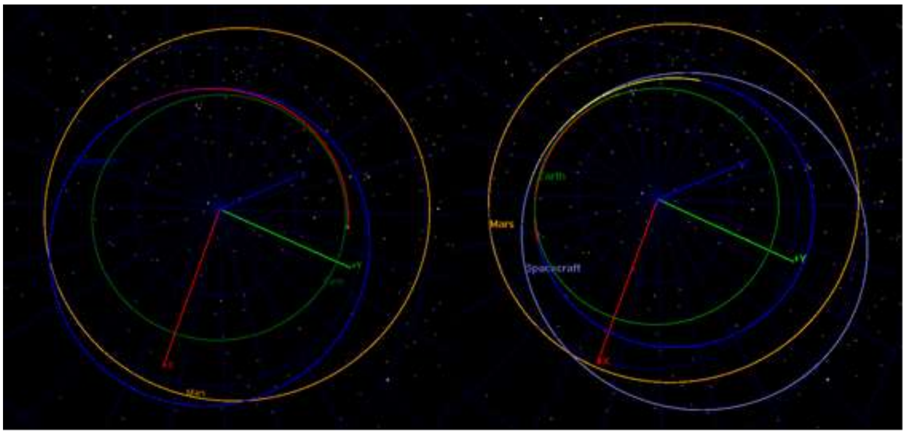

Figure 3: Single finite burn maneuver (Left). Double finite burn maneuver (Right).

Key Performance Parameters

The closest approach distance is set to 600 km, value resulting from an extensive analysis concerning different aspects of the mission and the main involved subsystems. This choice is motivated by the need to maintain the stabilization of the spacecraft due the impacting grains of the cometary environment. The attitude and orbit control subsystem guarantees the *three-axis stabilization* of the spacecraft with 3+1 redundant reaction wheels and a reaction control system. If the flyby velocity is too high or a reaction wheel fails, the closest approach distance can be increased to maintain an acceptable risk during the data collection. For a proper attitude determination and to satisfy pointing requirements two star trackers *(2 arcsec accuracy),* six sun sensors *(<0.5 deg accuracy)* and two inertial measurements units are required. In order to collect scientific data during flyby, the following main units are used: a multispectral camera *(2 m Ground Sampling Distance (GSD) at 400 km),* a spectrometer *(~70 m GSD at 400 km)* and a thermal camera *(~60 m GSD at 400 km).* These instruments are able to determine the nucleus characteristics and the chemical composition of the comet. To increase the collected data quality and quantity, the payload suite is completed with the following instruments: a mass spectrometer to determine cometary environment composition, three impact sensors and a three-axis fluxgate magnetometer to detect the interaction between the magnetic field of the comet and the solar wind. To transfer the collected data on ground, an X-band High Gain Antenna is used. Signal modulation is provided by the IRIS V2 X-band transponder unit. The power is amplified by means of a Travelling Wave Tube Amplifier (TWTA). The amount of data collectable and transmittable could potentially achieve 250 GB. All instruments have a high Technology Readiness Level (TRL) except for the electric thruster (TRL 5), which is an innovative technology. Its development process shall be carefully monitored to ensure the feasibility of the mission. Consequently, resources can be invested to bring this value to a high level at the end of the design phase.

Space Segment Description

Alternative mission concepts have been defined, changing the platform configuration and the payload embarked onboard. Following the System Engineering approach, extensive studies have been carried out to establish their main characteristics. A trade-off analysis has been performed through the Analytic Hierarchy Process[4] to determine which alternative best fulfils the mission objectives. Consequently, a functional baseline configuration has been designed. To ensure the compliance with mass constraints, a preliminary mass budget has been derived (Table 1). The mass allocation has been performed taking into account a 20% margin of the dry mass, according to the AIAA Recommended Weight Contingency Regulation. Similarly, a Power Budget (Table 1) has been obtained to guide the design phase of the Electric Power Subsystem. The sizing case for solar arrays is the *transfer phase* to ensure the required amount of power to the electrical thruster. Secondary batteries ensure the power during the lunar eclipses and peak power demands during the communication phase. In Table 2, delta-Vs for the different orbital design phases above discussed has been reported.

Mass Budget [kg]		
Structure	13.2	
Thermal Subsystem	0.810	
Attitude Control	6.86	
Power Subsystem	21.5	
Propulsion	4.13	
Communications/Data Handling	6.70	
Payload	7.5	
Total Dry Mass (20% margin)	72.8	
Propellant (chemical | electrical)	11.8	15.4
Total	100.0	

Power Budget [W]	Parking	Transfer	Flyby	Comm.
No Margin	83	325	108	168
Margin	45%			
Total	120	472	157	243

Link Budget			
Distance from Earth [AU]	0.5	1	1.5
Data Rate [kbps]	66	16.7	7.4
C/N_0 [dB]	53.7	47.7	44.2

Volume (without shield) [m^3]	0.102
Dimensions (stowed) [cm]	86.0 x 78.0 x 77.2

Table 1 - Mass Budget, Power Budget, Link Budget and dimensions.

Station Keeping Maneuvers	Departure from the Moon SOI	Departure from the Earth SOI	Double finite burn transfer
1 m/s per year	30-40 m/s	~200 m/s	~3250 m/s

Table 2 - Necessary delta-Vs for mission maneuvers.

In the structure mass is also included a shield against cometary dusts of about 3.5 kg. A Double Whipple Shield of 2.5 mm Kevlar and 0.3 mm aluminum with a 13 cm spacing between the two has been designed. The shield can protect the spacecraft from particles of 100 mg mass up to fly-by velocities of 64 km/s. In case the fly-by velocity is higher, the closest approach distance from the comet can be increased to maintain an acceptable risk of encountering critical particles. The shielding has been designed by resolving Ballistic Limit Equations through *ESABASE2*[5]. The shield is placed onto the three faces exposed to the ram direction during the flyby, when the satellite rotates to point the payload towards the comet. Costs have been analysed through SSCM and NICM models[6], from the mass of the subsystems and the power of the payload, resulting equal to about 35M$. This is a rough value obtained by simple models and shall be considered as a worst case. Through a more accurate cost analysis, the budget is expected to reduce. In addition, since the mission may be a module of the Lunar Gateway, its funding can be

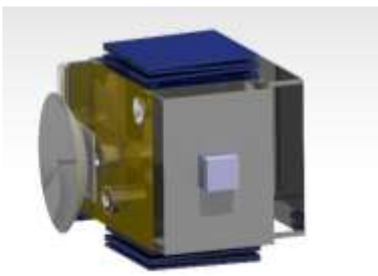

Figure 4 - CAD drawing of SCORE spacecraft (stowed).

connected to the Artemis Program[7], in which deep space missions deployed from the station are envisaged. In Figure 4 the CAD of the spacecraft is shown.

Additional considerations

The *flexibility* of the mission phases is crucial. Due to the probabilistic nature of the problem, phases are highly variable depending on the predicted position of the comet at the time of the flyby. The mission idea presented is a starting point for future mission analysis studies, proving the feasibility of satisfying the mission objectives with the limited resources available.

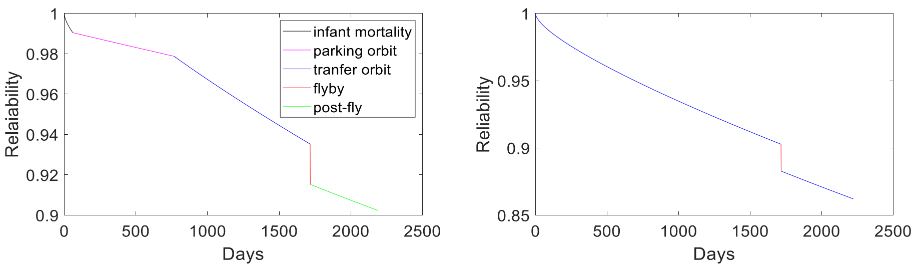

Figure 5: Reliability dividing the mission in different phases (left) and considering all subsystems active (right).

A reliability analysis has been performed using Weibull's distribution[8]. An example is reported in Figure 5 (left) by considering a specific mission scenario. In this case different parameters of the Weibull's distribution for each phase have been considered depending on the most used subsystem. In Figure 5 (right) the reliability has been calculated considering all subsystems active independently from mission phases. From this analysis upper and lower bounds of the reliability can be set, obtaining $0.86 < R < 0.90$. Since the shield against cometary dusts covers three faces of the spacecraft, accommodation solutions have been taken to correctly place all the elements of the system. In particular, the High Gain Antenna shall be positioned to avoid damages during the different mission phases. Depending on the configuration of the satellite (Figure 4), the only side free from the shield or other elements is the one where the thrusters are installed. A mechanism able to deploy the antenna is preliminarily designed to avoid its interaction with the thruster's plumes and to protect it from cometary dust during the fly-by.

References

[1] https://www.jpl.nasa.gov/cubesat/missions/marco.php (July 2021)

[2] https://ssd.jpl.nasa.gov/sbdb.cgi (June 2021)

[3] https://www.esa.int/Enabling_Support/Operations/ESA_Ground_Stations (June 2021)

[4] Saaty, T. L. (1990) "How to make a decision: The analytic hierarchy process", European Journal of Operational Research, Volume 48, Pages 9-26.

[5] https://esabase2.net/ (June 2021)

[6] J. R. Wertz , D.F. Everett, J.J. Puschell, "Space Mission Engineering: The New SMAD", Space Technology Series, Space Technology Library Vol.28, Springer, 2011.

[7] https://www.nasa.gov/artemisprogram (July 2021)

[6] Palla, C., Peroni, M., & Kingston, J. (2016). Failure analysis of satellite subsystems to define suitable de-orbit devices. Acta Astronautica, 128, 343-349.