Title: MELCHIOR: Micro-satellite Explorer to a Long-period Comet in a Heliocentric Inner ORbit

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Mission Objectives (where and why?)

Comets are the most primitive objects in the Solar System. These celestial bodies have kept a record of the physical and chemical processes that occurred during the early stages of the evolution of our Solar System.
Studying Long Period Comets (LPC) is of great interest to the scientific community, since these pristine objects has experienced only limited Sun alteration and they are only slightly polluted by solar radiation. Furthermore, LPCs come from regions that cannot be reached by ground telescopes so, thanks to the measurement the mission is projected to perform, it is possible to investigate about the validity of the Oort Cloud model, the stationary region where these celestial bodies are scattered due to a gravitational perturbation.
MELCHIOR is a mission aiming to characterize surface, shape, structure and chemical composition of an LPC, seeking clues of organic material such as carbon and amine, which are at the basis of the DNA molecules. It embarks three optical sensors, a three-axis flux-gate magnetometer, a dust impact sensor and a mass spectrometer (see Section Key Performance Parameters). The mission starts with a spacecraft delivered in a southern Moon-Earth L2 resonant Near Rectilinear Halo Orbit (NRHO) e.g. from the upcoming Lunar Gateway, with a perilune radius of 5931km, waiting in that orbit until a new LPC is discovered. As a result of mission analysis and preliminary subsystem design, the mission is conceived to perform a fly-by with an LPC crossing the ecliptic plane in a Sun-centered annulus with a radius between 0.87 and 1 Astronomical Unit.
MELCHIOR shall be considered a one-of-a-kind mission, since an encounter with an LPC would provide valuable data to complement those supplied by previous cometary missions, which were limited to short-period comets. On the other hand, given the probability that no suitable target might be identified, a short-period comet, MIDAS 1981 is selected as backup target to guarantee a meaningful scientific return.

Concept of Operations including orbital design

The unpredictability of this kind of target has been the main driver for the identification of the different mission phases (Figure 1). The mission starts before 2030 with the spacecraft on the parking orbit, selected for its stability (less than 7 m/s as annual ΔV for station keeping) and eclipses avoidance [1].

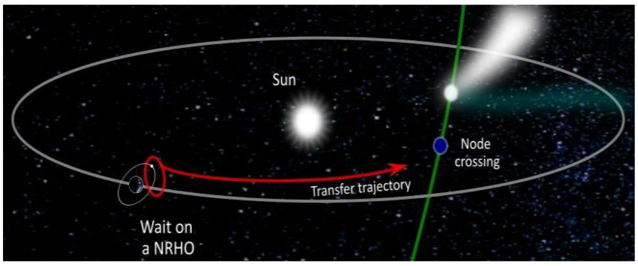

Figure 1 – *Sequence of mission phases*

The waiting phase is assumed to last up to 6 years. Once the discovery of a new possible target is reported from ground, data related to its trajectory are computed so that the transfer phase can be planned. The encounter shall take place at the ascending or descending node to remain on the ecliptic plane while saving propellant. Moon position is also important to further minimize the required ΔV. As a result, the satellite can be asked to wait up to 23 days to reach a favorable configuration for the departure from the NRHO. Additionally, considering the very small value of the heliocentric inclination when the spacecraft is at the perilune of the parking orbit, only tangential $\Delta V s$ are needed. The problem is parametrized in terms of distance from the Sun at encounter (R_C), and Earth-Sun-Comet angle at encounter (ϑ). A double impulsive maneuver strategy is taken into account: an example of the trajectory obtained using STK software is reported in Figure 2. Because of constraints in terms of mass, a maximum ΔV of 1000 m/s is reserved to the transfer, with no longer than 3.5 years of transfer time. With these numbers, the reachable area, in terms of points on the ecliptic plane that the spacecraft can have access to, can be obtained (Figure 3). When the satellite is at a distance of 100.000 km from the target, the fly-by phase starts, and optical instruments are turned on. At a distance of 50.000 km, the remaining sensors are switched on. The actual duration of the fly-by is affected by the inclination of the comet with respect to the ecliptic plane, with several tens of minutes in the worst case (Figure 4). The impulsive maneuvers are designed to place the spacecraft between the Sun and the target at the moment of the closest approach, to avoid having the Sun in the field of view. During this amount of time, attitude maneuvers have to be performed in order to continuously point the payload instruments towards the comet. Therefore, here the power is provided by a secondary battery to compensate an eventual off-nominal pointing of the solar panels. As for the post-encounter phase, it is required to transmit all the collected data to the ground. Taking into account a prediction of the amount of data to download, 8 h/day of communication with the ground stations, and distance from the Earth, the link budget analysis confirmed that the post encounter phase shall last up to 9 months.

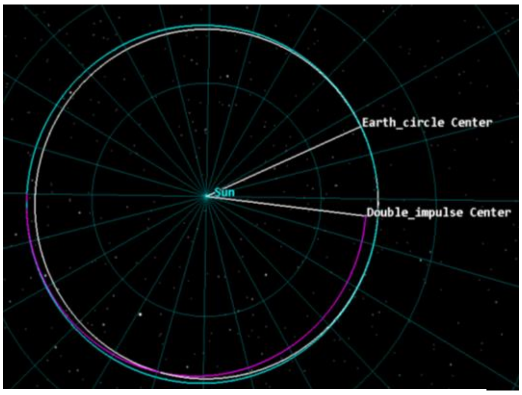

Figure 2 - *Double impulsive maneuver (R_C=0.95 AU, ϑ=-30°)*

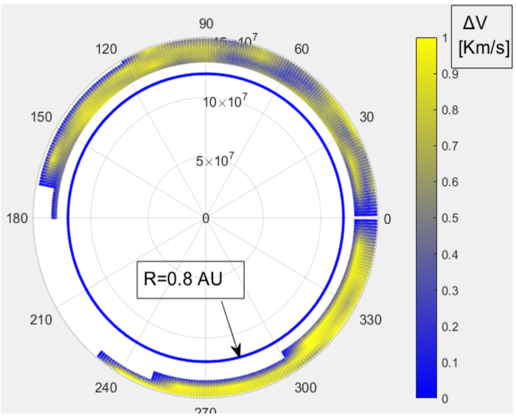

Figure 3 - *Reachable area (R_C, ϑ)*

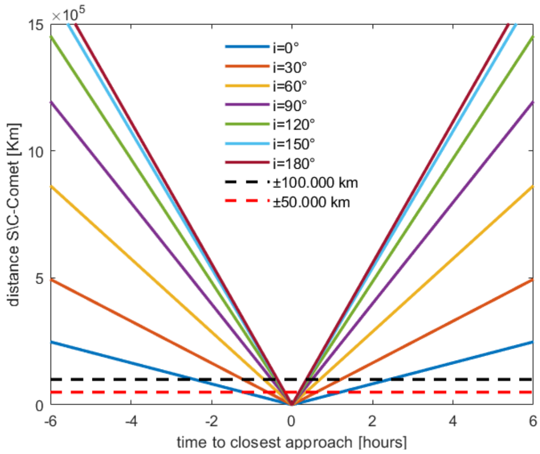

Figure 4 - *Expected fly-by duration for different comet inclinations*

Key Performance Parameters

Starting from mission objectives and LPC characteristics, different sensors are embarked:

- Multispectral camera to analyze the surface of the nucleus, with a spatial resolution of 14.25 m/pixel at a distance of 1000 km;
- Short Wave Infrared and Long Wave Infrared spectrometers to map chemical composition of nucleus and coma and surface temperature of the nucleus, with a spectral resolution of 0.02 µm and 0.2 µm respectively;
- Mass Spectrometers to analyze dust particles composition, with a mass resolution of 10-12;
- Dust impact sensor to study dynamical properties of cometary dust ejected by the nucleus;
- Flux-gate magnetometer to study the interaction of solar wind plasma with coma.

The Attitude Control Subsystem is designed to guarantee an accurate pointing of payload instruments during the fly-by phase. The attitude determination task is entrusted to star trackers and Sun sensors, with an accuracy of a hundredth of degree and Inertial Measurement Unit with a gyro resolution of 0.22°/h. After the definition of moments of inertia and maximum angular velocity needed to follow the comet path during the fly-by, four reaction wheels in a pyramidal configuration have been sized to perform attitude control: the needed maximum angular momentum is estimated to about 0.34 Nms. Concerning the propulsion

system, different configurations have been considered. A single electric propeller has been discarded for the impossibility of escaping from the Moon sphere of influence, as well as a hybrid configuration (electric + chemical) has been rejected to respect mass constraints. Therefore, the baseline concept is equipped with a single chemical propeller. Orbit control is provided by means of four 1N hydrazine thrusters, while station keeping maneuvers and transfer are guaranteed by a Dual Mode thruster with a specific impulse of 329 s and a nominal thrust of 400 N. About 200W are needed for on board instruments: these are provided by means of 2 deployable solar panels composed of quadruple junction solar cells, to guarantee an efficiency at beginning of life of 32%. A secondary battery of 512 Wh is embarked to provide power also during the fly-by phase. Distribution is then ensured by a power conditioning and distribution unit, with efficiency of energy transfer larger than 98%. To minimize the threat of failure or perforation due to impacts with micrometeoroid, SPENVIS software analysis highlighted the necessity of a stuffed Whipple shielding for the three out of six faces exposed to cometary dust, with a 2.25 mm-Kevlar 49 layer inserted between the Aluminum outer bumper and the rear wall. Instead, shielding against the radiation environment is guaranteed by the Aluminum box structure. Since the avoidance of eclipses events is guaranteed, thermal control is supplied only by passive systems as a 2mm-Aluminzed Kapton protective coating on the spacecraft surfaces.
Telecommunication with NASA DSN [2] is provided by the reflectarray High Gain (>29dB) Antenna [3], with a 65 W Travelling Wave Tube Amplifier and the NASA Small Deep Space Transponder selected for its deep space heritage, long lifetime and capability of providing the spacecraft with two-way Doppler link. This solution ensures download of the maximum amount of collectable scientific data (42Gbit with fly-by duration of 2.6h) within 9 months for the worst post-encounter trajectory that can reach up to 1.5 AU from the Earth. In the parking phase, the Low Gain Antenna and a 5W amplifier are enough for common TT&C operations: in each link, with a bit-energy to noise ratio of 3.4 dB, selecting a proper modulation method, a Bit Error Rate lower than 10^{-5} is guaranteed.

Space Segment Description

The space segment has been preliminary designed based on suitable off-the-shelf components compliant with subsystem requirements. Components and their properties are listed in Table 1, where a 20% margin is included for the dry mass, whose total value takes into account also the propellant contribution (30 kg).

	Component	Qty.	Mass [kg]	Dimensions [mm³]	Max Power [W]	TRL
Payload	Camera: Multiscape 100 CIS	1	1.4	98 x 98 x 176	5.8	7
	LWIR: TAU 640 100 mm f/1.6	1	0.48	44 x 44 x 44	1.2	6
	SWIR: TAU	1	0.17	38 x 38 x 36	3.2	6
	Mass spectrometer: INMS	1	0.56	90 x 100 x 130	1.8	6
	Dust impact sensor: DISC	1	0.40	125 x 120 x 130	2	5
	Magnetometer	1	0.1	90 x 100 x 82	N/A	6
Shielding	Box structure	1	5.83	600 x 600 x 1	-	-
	Stuffed Whipple (Aluminium)	3	2.92	600 x 600 x 1	-	-
	Stuffed Whipple (Kevlar 49)	3	3.5	600 x 600 x 2.25	-	-
TCS	Aluminized Kapton Coating	-	0.5	2mm of thickness	-	8
EPS	Solar panels	2	0.66	720 x 720 x 40	-	6
	PCDU	1	2.5	395 x 125 x 65	-	6
	Secondary battery	1	5	308 x 180 x 90	-	7
	Deployment mechanism	4	6	122 x 110 x 70	-	6
	Hold-Down and Release Mechanism	4	0.08	18.7 x 38.45 x 5.7	-	6
ACS	Actuators: RW 90 – Astrofein	4	3.6	Ø103 x Ø101 x 60	14	6
	Star sensor: CubeStar CubeSpace	2	0.11	50 x 35 x 1.61	0.28	6
	Sun sensor: NCSS-SA05	6	0.03	33 x 11 x 6	-	6
	Ch. Thruster: 1 N Ariane Group	4	1.16	Ø 50 x 172	-	6
	IMU: STIM 300	2	0.11	39 x 45 x 22	1.2	6

Propulsion	Thruster: R-4D-15 HiPAT	1	5.44	726 x 362	-	6
	Tank: 80608-1 Northrop Grumman	2	4.69	Ø 117.5	-	6
	Shielding for Tank	2	0.33	Thickness 0.216	-	6
TLC	Small Deep Space Transponder	1	3.2	180 x 165 x 114	16	9
	SDR: TOTEM Alèn Space	1	0.13	93 x 89 x 5	2	6
	MarCO High Gain Antenna	1	1	597 x 335 x 4	-	6
	4x2 microstrip Patch antenna	4	0.4	92 x 42 x 4.7	-	-
	COBRA-HPX pointing mechanism	1	0.3	Ø 113 x 29	2.5	6
	X-band Patch Antenna Endurosat	4	0.1	24 x 24 x 6	-	6
	5W Qorvo power amplifier TGA2701	1	0.01	5 x 5 x 1	8.5	6
	65W Travelling Wave Tube Amplifier	1	0.8	70 x 50 x 220	115	7
	Switch/Filters/Diplexers	1	1	N/A	-	-
Total			**99.88**		**195.02**	

Table 1 - *MELCHIOR equipment list, mass budget and power budget*

A 3D model of the spacecraft has been developed (Figure 5) to provide each equipment reported in Table 1 with its possible accommodation (Figure 6), to obtain more accurate moments of inertia for ACS preliminary design and to preliminarily confirm that the volume constraint is met.

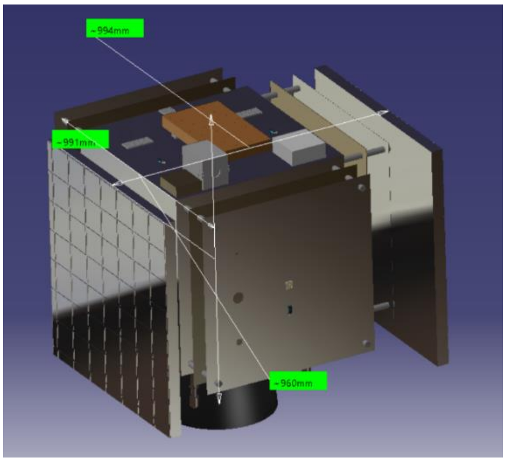

Figure 5 – *Spacecraft in its stowed configuration with envelope size*

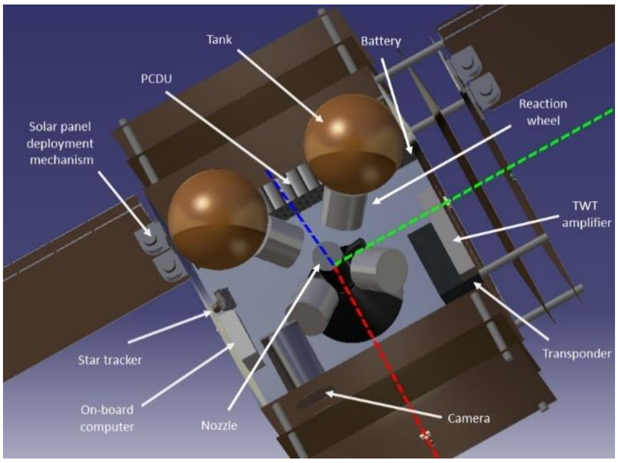

Figure 6 - *Equipment internal allocation*

Additional considerations

The mission under analysis has to deal with the unpredictability of the target. The limited number of LPCs detected per year, jointly with the restricted reachable area in the heliocentric annulus and the randomness of timespan between LPC detection and perihelion passage, have been considered in a probabilistic study to estimate the probability that at least one reachable target is detected during the parking phase. Given that the heliocentric distance at nodes, fixing eccentricity at a value of 0.99, is a function of just 2 orbital parameters (radius and anomaly of perihelion, r_p and ω respectively), starting from data about past LPCs collected in [4], the two-variables joint Probability Density Function has been estimated and integrated under the region 0.8 – 1 AU, obtaining the probability that a detected comet will pass in this annulus i.e. P(A) = 0.05, as shown in Figure 7. R_C and ϑ are assumed to be random variables

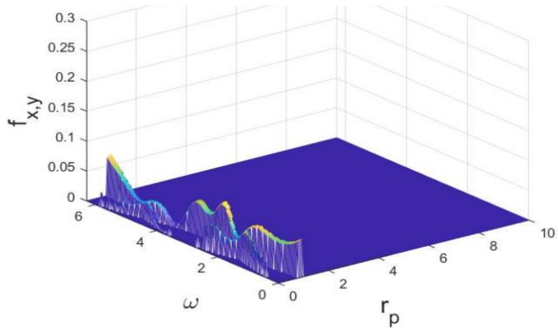

Figure 7 - *Joint PDF of Sun distance at nodes restricted to the region 0.8 - 1 AU*

with uniform distribution: it represents a reasonable approximation regarding the former due to the small range that is considered (0.8 – 1 AU) and about the latter because it depends on comet passing epoch and its right ascension of ascending node (whose CDF is well-approximated by a linear function, as visible in [5]). So, the probability that a comet passing between 0.8 – 1 AU also requires a ΔV lower than the maximum one, is estimated by the ratio between the reachable area shown in Figure 3 and the whole annulus i.e. P(B|A) = 0.46. Also, in a conservative approach, the percentage of comets for which time between detection and perihelion passage (distributed as in [6]) is lower than maximum maneuver time (3.5 years) is neglected i.e. P(C|A,B) = 0.70. Finally, with an expected value of 24 LPCs detected/year during the mission [2], the probability of having at least 1 reachable LPC during the parking phase, P_{LPC}, was obtained by means of a binomial random variable, as shown in Table 2.

P(A)	P(B|A)	P(C|A,B)	P(A∩B∩C)	P_{LPC}
0.05	0.46	0.70	0.016	0.906

Table 2 - *Results of probabilistic study*

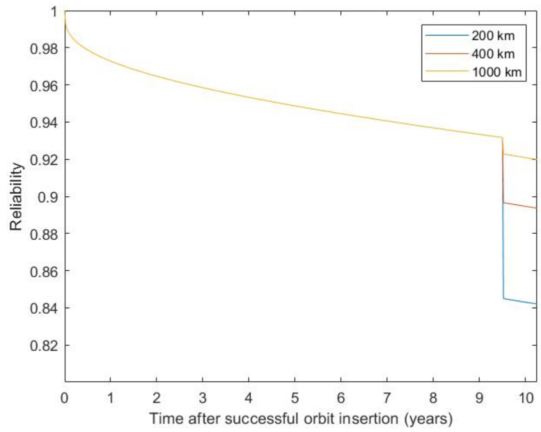

Figure 8 – *System reliability for three different distances of closest approach*

The reliability of the system has been also evaluated to analyse the feasibility of the mission: Weibull distributions for all the subsystems have been considered [7] and then, to obtain values at system level, the reliability of each subsystem has been multiplied as in a series connection. Other factors considered are the operational duty cycle in each one of the mission phases and the probability of impact with comet dust particles for three different distances of closest approach. In Figure 8 results are shown: only the distance of 1000 km guarantee a reliability of over 90% for the entire lifetime. Hence, the distance of the closest approach is set to 1000 km in the baseline solution. Two models have been used to estimate the total cost of MELCHIOR mission: Small Satellite Cost Model and NASA Instrument Cost Model [8]. The costs reported in Table 3 are in fiscal year 2021 and comprehend costs for payload, spacecraft bus, integration, assembly and test, program level, launch and orbital operations support, aerospace ground equipment, flight software, phase A, ground segment and years of mission operations. Best case and worst case have been obtained considering the standard error of the estimate of the models.

Best Case [$M]	Average Value [$M]	Worst Case [$M]
77.931	106.381	134.811

Table 3 - *Cost estimation by Small Satellite and NASA Instrument Cost Models*

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