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H.E.R.M.E.S.: A CUBESAT BASED CONSTELLATION FOR THE NEW GENERATION OF MULTI-MESSENGER ASTROPHYSICS

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H.E.R.M.E.S., a mission to fly in 2022, is the first 6 high performance 3U CubeSats LEO constellation to collect multi-messenger astrophysics data devoted through a fractionated payload strategy.

The paper describes the Mission Analysis performed to achieve the highest possible sky coverage Sky throughout the mission duration, exploiting an innovative tool to optimize the coordinated pointing strategy of the distinct space elements, coupled with their natural relative orbital dynamics.

A wide coverage analysis and communication passages optimization is discussed aiming to minimize the latency between the astrophysical event detection and its ground communication.

INTRODUCTION

The High Energy Rapid Modular Ensemble of Satellites, H.E.R.M.E.S., is a challenging scientific space mission which aims contributing to the new Multi-Messenger Astrophysics , by cleverly distributing a set of sensors on orbit, to timely localize Gamma Rays Bursts (GRB), trace of Gravitational Waves generation, while continuously monitor the celestial sphere. Six novel miniaturized X and Gamma rays' detectors are installed on a dedicated 3U CubeSat to form the core of a quasi-Equatorial, Low Earth Orbits constellation.[1] These multiple space assets, perform coordinated sky monitoring and localization through triangularization, being the implementation of a fractionated large detector. Sky monitoring shall be the widest possible and a timely transfer on ground of the cosmic event localization coordinates, whenever occurring, is mandatory (order of magnitude: 15 min), to allow powerful Earth-based instrumentation to investigate the more the detected relevant sky area.

H.E.R.M.E.S., to fly in year 2022, is a project co-founded by the National Ministry for Research (MUR), the Italian Space Agency, and the European Union's Horizon 2020 research and innovation program. Politecnico di Milano, Department of Aerospace Science and Technology, ASTRA research group, together with INAF, oversees the space segments and payload implementation, respectively.

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During operations, the Field of View (FOV) of at least three instruments, separated by a minimum physical baseline of 1000 km, shall overlap to observe a common sky area to ensure the triangularization feasibility in case of random cosmic events occurrence. Since the triangulation leans on the time delay between the three incoming signals by pairs, the scientific desired delay of 3 ms is achieved whenever at least two projected baselines (i.e. the physical baseline projected in the direction of the payload's pointing direction) are larger than 1000 km.[2] Therefore, the satisfaction of very challenging scientific requirements can be met only smartly tuning the mission analysis degrees of freedom offered by the six space segments orbital and attitude relative dynamics. The CubeSat class selected for the service modules, imposes not negligible constraints such as the lack of on-board translational control authority and the limited on-board computational and power resources.[3] Last but not least, to fly as piggybacks, with no dedicated launch, introduces relevant uncertainties on injection into orbit conditions the constellation shall be robust against.

The paper critically discusses the mission analysis performed to achieve the constellation highest possible sky visibility throughout the mission duration, nominally 2 years. The proposed methodology combines a high-fidelity orbit propagator with an innovative mission analysis tool that can estimate the scientific performances of the constellation, with a parametrized number of space assets.[4] The optimization strategies settled to compute the six coordinated pointing directions – by variable triplets of satellites – is presented. The capability to obtain a flexible pointing profile along the mission timeline, is a powerful tool to maximize the scientific return during H.E.R.M.E.S. constellation operations, attainable just uploading a new payload pointing plan, whenever required. The discussion stresses that, being orbital correction maneuvers unavailable on-board, a deterministic approach has been discarded: the Mission Analysis sizing, to get to a robust solution, asked for a wide statistical campaign, to include all possible uncertainties affecting the orbits and the injection conditions; both nominal and non-nominal scenarios have been assessed.

H.E.R.M.E.S. has a point of strength in its fractionated design which, on the other end, makes the free relative dynamics leading the constellation performance along time and needs to be properly tuned playing with that dynamic's initial conditions,[5] being those the only control vector available. The injection strategy to achieve and naturally bound the physical baseline is presented in the paper. The fundamental idea is to have simultaneously released triplets, to get the desired baseline occurring after few days with a periodic evolution of in the order of $10^2[d]$. Thus, to assess the injection strategy to guarantee the fulfillment of mission objective is crucial.

The H.E.R.M.E.S. scientific objectives impose stringent requirements on the data transfer to rapidly inform the Scientific Operation Center about the triangulated event. To this end, the mission baseline includes both Ground Stations and Space Relay Networks exploitation, the visibility of which further stretched the Mission Analysis definition. Results discussed in the paper will highlight the compliance of the settled baseline to the gap between contacts minimization, whenever the astrophysical event is detected and localized.

The presented methodology can be easily extended to any kind of distributed scientific space applications, as well as to constellations dedicated to Earth and planetary observation. Even if the current H.E.R.M.E.S. fleet limits to six units, a further evolution of its space segment is foreseen to get to the continuous sky coverage; results about the effectiveness of the constellation enlargement by inserting new units on different orbital planes, with focus on highly inclined opportunities, are part of the paper content.

MISSION SCENARIO BASELINE

The definition of the mission scenario baseline, starting from the scientific requirements, is intended to select:

- the nominal orbit of the spacecrafts;
- the injection strategy of the space segment;
- the pointing strategy of the payload's line of sight.

The output of the mission analysis is selected according to the estimates of the scientific performances. A key performance parameter is defined in order to evaluate the scientific return of the mission. The logic driving the mission analysis is synthesized in Fig. 1.

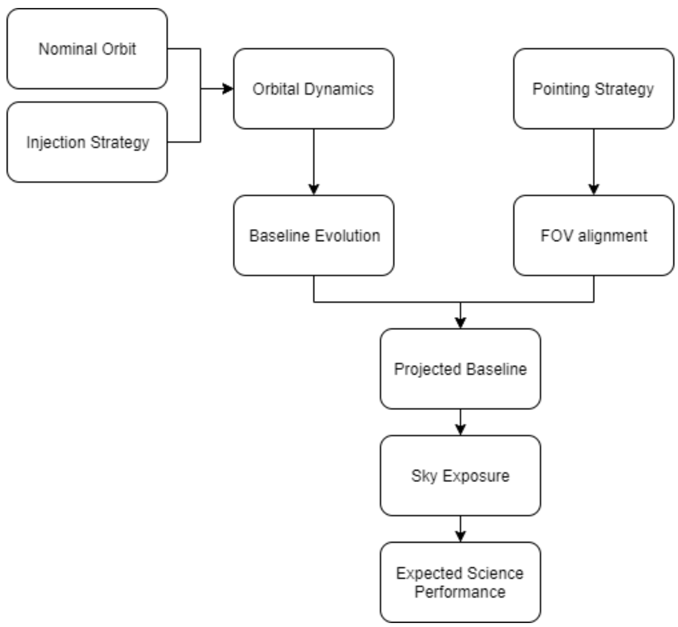

Figure 1: Flowchart of the logic driving the mission analysis.

The operational orbit and the injection strategy determine the natural dynamics of the system and the evolution of the relative orbits of the spacecrafts. On the other hand, the pointing strategy is crucial to perform the FOV alignment. The FOV alignment together with the baseline evolution yields the projected baseline, which is relevant to assess the fulfilment of the scientific objective. The projected baseline allows to determine the region of the sky that can be triangulated and hence the expected scientific outcome in terms of number of triangulated GRBs. It shall be noted how the lack of orbital control requires a careful modeling of the relative dynamics from the injection, with all the possible mission uncertainties. The unavoidable relative drift is overcome by exploiting the advanced attitude control operations, that are designed to have frequent re-alignment of the drifted FOV due to the natural dynamics.

The scientific requirements applied during the mission analysis and orbit design are:

- The altitude of the operational orbit shall be lower than $600\,\mathrm{km}$;

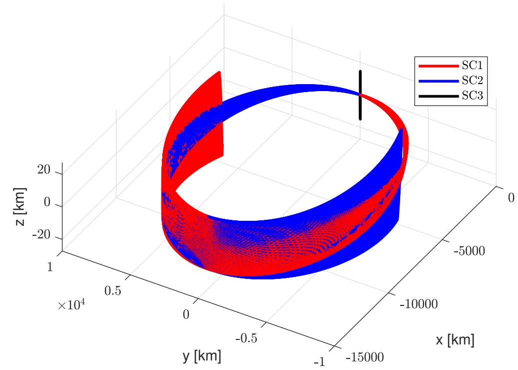

Figure 2: Relative motion of a triplet after injection on +t-bar, -t-bar and h-bar

- The inclination of the operational orbit shall be lower than $30°$ or greater than $70°$;
- The mission duration shall be greater than $2\,\mathrm{y}$.

Operational Orbit

The operational orbit selection is the first output of the mission analysis. The application of the previously introduced scientific requirements constrains the orbit design to:

- Equatorial or nearly-equatorial orbit with $450\,\mathrm{km} \leq h \leq 600\,\mathrm{km}$;
- Sun Synchronous orbit (SSO) with $450\,\mathrm{km} \leq h \leq 600\,\mathrm{km}$.

Both orbital families are feasible with commercial launchers. The lower bound for the altitude boundary comes from the mission duration requirement. In fact, a spacecraft like H.E.R.M.E.S. with a mass to area ratio in the order of $30\,\mathrm{kgm}^{-}2$ has an orbital lifetime greater than $2\,\mathrm{y}$ for an initial orbit altitude greater than $450\,\mathrm{km}$, with average solar activity (i.e. $F_{10.7} \sim 110\,\mathrm{sfu}$).

Low Earth Equatorial Orbits are preferred as a result of a performed trade-off with the payload requirements, since the scientific instrument degradation that would occur at high inclinations is considered too risky for the pathfinder mission design. Sun synchronous orbits, or other high inclination orbits, may be relevant for a future extension of the current mission design.

Constellation Injection

The constellation injection is designed to achieve the required physical baseline of $1000\,\mathrm{km}$ by exploiting an imposed natural motion of the satellites. This is mainly due to overcome the lack of orbit control capabilities. In fact, a precise relative dynamics control is extremely complex to be achieved in these conditions. Moreover, the simultaneous injection of the constellation satisfies the typical launch constraints impacting nano-satellites.

In order to impose a safe relative motion that separates the spacecraft, the spring injection impulse is exploited. In particular, the injection strategy features three orthogonal directions, one for each satellite of the triplet. The reference frame is the Local-Vertical-Local-Horizontal (LVLH), which is defined by the radial vector pointing outward r-bar (i.e. zenith direction) and by the orbital angular momentum h-bar. The third axis, t-bar, completes the right-handed triad, in the horizontal direction with respect to the ground. For circular orbits, the t-bar is coincident with the velocity vector. The first satellite is inserted into a cross-track harmonic motion around the injection anomaly, thus bounded in the LVLH frame centered in the aforementioned anomaly. The two remaining satellites are inserted into a drifting orbit along +t-bar and -t-bar, respectively, as shown in Fig.2. The drifting component is necessary to achieve the large required baseline.

This strategy is actually beneficial with respect to a dedicated injection for each satellite. Indeed, the imposed relative motion is governed by the spring impulse, thus under design control. On the other hand, dedicated injections, i.e. each satellite inserted at a predetermined anomaly, still holds a drifting motion, which is in turn highly sensitive to the true injection anomalies and to the launcher uncertainties. The true injection anomaly is hard to be precisely predicted, hence the resultant motion is highly uncertain and cannot be regarded as a controllable design variable.

Pointing Directions

The selection of the nominal pointing strategy for the scientific mode significantly affects the H.E.R.M.E.S. performances. Albeit H.E.R.M.E.S. satellites cannot perform orbit control, their attitude control performances are quite advanced. The capabilities offered by the attitude control can be exploited to enhance the whole mission scientific return.

In fact, the proposed constellation injection strategy results in a continuous relative drift between the different space elements. The attitude control is exploited to avoid the consequent drift of the overlapped FOV, which would result in a detriment of the Sky area available to perform triangulation of the incoming cosmic event. The alignment of the payloads' line of sight (LOS) is performed by the attitude control system, which has been designed to frequently maneuver the spacecrafts.[3] The alignment directions are computed on ground, according to different possible pointing strategies, and uploaded on-board in the form of quaternion profiles. The alignment maneuvers are performed with a certain frequency, resulting in a periodic update of the pointing directions. An update frequency of 1 week resulted to provide a good compromise between control effort and scientific results.[4]

The pointing direction strategies considered for H.E.R.M.E.S. are:

- Co-alignment of $n \geq 3$ payload LoSs on a LVLH-selected direction (i.e. LoSs aligned on the zenith direction of a specific satellite in the fleet, selected on-ground).
- Co-alignment of $n \geq 3$ payload LoSs on a LVLH-optimized direction (i.e. LoSs aligned on a optimal direction, defined in the LVLH frame, selected and computed on-ground).

LVLH-selected direction co-alignment The selected operational scenario for the co-alignment of $n \geq 3$ payload LoSs on a LVLH-selected direction considers 6 satellites in orbit; for any scientific operation time segment (e.g. 1 week) two logically separated triplets are defined (i.e. triplet n.1 and triplet n.2), according to the actual position of the spacecrafts in orbit. Then, the pointing strategy is defined for both triplets: a *reference spacecraft* is selected from ground, for each scientific operation time segment. The *reference spacecraft* is aligned with its zenith direction and the other two spacecrafts in the triplet are controlled to align their FoVs to the *reference* one. The spacecraft in the triplet selected as reference is not fixed in the time, but varies for each time segment. The output of the analyses shows that the scientific results are almost independent from the *reference element* selection.

This pointing strategy simplifies and reduces the maneuvers to be dedicated for the scientific operations. However, this pointing strategy is prone to have a portion of the Earth sphere within the FoV for a certain amount of time. Moreover, the LVLH references direction may be affected by passages of the LoSs over sky regions to be avoided, according to the scientific measurements plan (e.g. galactic bulge). Fig. 3 shows the pointing strategy with alignment on a LVLH-selected direction at an instant in time during the mission.

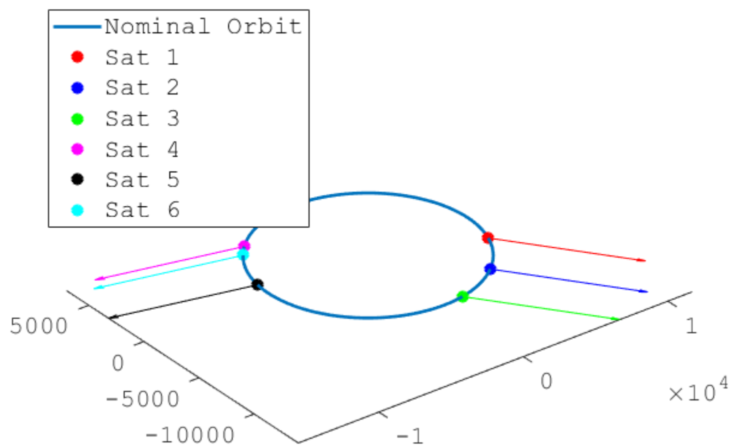

Figure 3: Co-alignment on a LVLH-selected direction.

LVLH-optimized direction co-alignment The selected operational scenario for the co-alignment of $n \geq 3$ payload LoSs on a LVLH-optimized direction is set as in the previous case. In analogy, the pointing strategy is defined for both triplets: a *reference LVLH directions* is selected by optimization with ground-based flight dynamics analysis, for each scientific operation time segment (e.g. 1 week). The three spacecrafts in the triplet are controlled to align their FoVs to the *reference* direction.

The reference direction is selected as the one maximizing the scientific outcome of the mission. In fact, this pointing strategy is defined to increase the useful overlapped FOV area, exclude the Earth sphere from the FoV and achieve the optimal disposition of the LOSs for each operational time segment. In fact, the optimization procedure can be set to also minimize the passages of the LoSs over sky regions to be avoided. However, it requires dedicated attitude maneuvers to orient every spacecrafts and to maintain any reference pointing direction along the time segment. Fig. 4 shows the pointing strategy with alignment on a LVLH-optimized direction at an instant in time during the mission. The details of the optimization with ground-based flight dynamics analysis are available in.[4]

SCIENTIFIC PERFORMANCES

The scientific performances evaluation and the mission analysis is performed with a dedicated utility, part of the high-fidelity orbital propagator developed at Politecnico di Milano by the MAT.

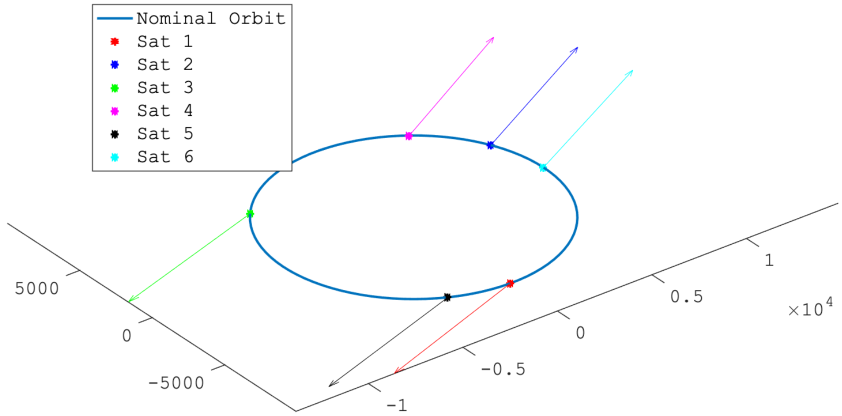

Figure 4: Co-alignment on a LVLH-optimized direction.

The developed analysis software tool adopted to evaluate the scientific performances of H.E.R.M.E.S. fleet has two main blocks:

- Orbital Propagator.
- Visibility Tool.

The Orbital Propagator outputs the time evolution of each spacecraft state vector with 1 minute time discretization as upper threshold. The orbital propagator takes into account all relevant perturbing forces acting in the operational orbits, namely:

- Earth geopotential based on EGM96 model. The accepted truncation error for the spherical harmonics order at the given altitude is selected according to the intrinsic accuracy of the EGM96 model, as per ECSS guidelines.[6]
- Solar gravitational perturbation, with Sun's position based on DE431 Ephemeris model.
- Lunar gravitational perturbation, with Moon's position based on DE431 Ephemeris model.
- Atmospheric drag perturbation on flat surfaces. The atmosphere density is based on Jacchia-Roberts model. Solar activity coefficients (F107-AP index) are available from NOAA Solar flux prediction model - 2016 update.
- Solar Radiation Pressure (SRP) on flat surfaces. Solar activity is taken into account and the solar coefficients (F107-AP index) are available from NOAA Solar flux prediction model - 2016 update.

The Orbital Propagator is an internally developed numerical tool validated with 3rd party orbital propagator software.

The visibility tool takes as input the trajectory of the n satellites computed through the orbital propagator and evaluates the expected number of GRBs to triangulate during the entire mission.[4] It simulates the projection of the spacecrafts FOV over the celestial sphere, implementing all the scientific requirements and constraints. Moreover, it considers the pointing definition and the attitude states of the spacecrafts.

Scientific performances are evaluated according to the performance index of sky visibility achieved by the H.E.R.M.E.S. fleet. The performance index takes into account the area of the sky that is visible and triangulable, while satisfying all imposed scientific requirements and constraints to the H.E.R.M.E.S. satellites; the triangulable sky area availability in time is part of the performance index as well. The resulting performance index is defined as an area of the sky that can be view for a certain amount of time and, thus, is denoted as time-area parameter: tA, whose dimensional units are steradians - day $[\mathrm{sr}-\mathrm{d}]$.

Time-area parameter is defined so to be well suited to estimate the number of expected GRBs detected by the H.E.R.M.E.S. fleet. In fact, by assuming the GRB detection rate of FERMI-GBM instrument (e.g. $0.083\,\mathrm{GRB/sr-d}$) as an estimate for the amount of detectable GRBs,[7,8] the tA parameter is easily converted into an expected number of detected GRBs.

Nominal Mission Scenario

The nominal mission scenario is defined in terms of nominal orbit, constellation injection and pointing strategy. Further, the last element to determine is the best nominal separation between the two triplets (i.e. the nominal true anomaly phasing required to the launcher while launching the two triplets). This is needed to completely define the selected injection strategy. The best conditions are achieved when the two triplets are injected with an angular shift, $\Delta\theta_0$ in the order of $140°$ - $230°$. Namely, the central elements (i.e. the one injected along the cross-track direction) are placed around the opposite sides of the orbit. The nominal mission scenario is characterized by the nominal values described in Tab. 1.

Table 1: Nominal mission scenario

	Nominal Value
Launcher	
Launch Day [y]	01/04/2021
Phasing [d]	1
SMA [km]	6928
e [-]	0
i [deg]	0
ω [deg]	0
Ω [deg]	0
θ [deg]	0 and 220
Release	
Spring $\Delta v[m/s]$	1.25

Once the nominal mission scenario is completely set, including the angular shift between the triplets, a comprehensive statistical analysis has been carried out to assess the robustness of the proposed mission scenario and the confidence on expected scientific return (i.e. number of GRBs), as a function of the unavoidable injection uncertainties. The pool of uncertain variables adopted for the robustness analysis is reported in Tab. 2. The selected set includes launcher's and deployer's injection errors. The number of runs and the guidelines to set-up the Monte Carlo statistical analysis are taken from the NASA technical report on statistical analysis for space missions.[9]

The reference values for launcher injection uncertainties come from the combination of the worst cases among the European launchers (Vega, Ariane 5, Soyuz). Furthermore, all the injection uncertainties are margined by 100%. After each triplet is injected by the launcher with uncertainties into the orbit, the spring release of each satellite from the deployer is taken into account. The spring release is affected by uncertainty in both the magnitude and direction of the spring Δv, as listed in Tab. 2. The injection angle uncertainty is taken from the worst cases among the European launchers in terms of upper stage attitude control accuracy, margined by 100%. For what concern the uncertainty on the magnitude of the momentum variation imposed by the release spring, it is imposed as uniform distribution around the design nominal value: $\Delta v_{nom} = 1,25 m/s$. Moreover, each spacecraft is injected with an individual release Δv, distributed normally with $10\,\%$ range at 3σ.

The goal of the Monte Carlo analysis is to estimate the value of the expected scientific outcome, in terms of GRBs seen by H.E.R.M.E.S. In terms of statistical analysis, the objective is to compute the mean value and to bound the scientific output within a box at $\sim 99.73\%$ of probability.

Table 2: Variables ranges and uncertainties

	Range		Uncertainty	
	Value	Distr.	3σ	Distr.
Launcher				
Launch Window [y]	2	Uniform	-	-
Phasing [h]	24	Uniform	-	-
SMA [km]	-	-	30	Normal
e [-]	-	-	0.0024	Half-normal
i [deg]	-	-	0.30	Normal
ω [deg]	-	-	1.2	Normal
Ω [deg]	-	-	0.4	Normal
θ [deg]	-	-	1.2	Normal
Release				
Spring Δv [$\frac{m}{s}$]	0.25	Uniform	10% of Δv_{nom}	Normal
Injection angle [deg]	-	-	4	Normal

Co-alignment on LVLH-selected directions The scenario with pointing strategy defined as co-alignment on LVLH-selected directions is characterized by the robustness analysis results in Fig. 5, 6 and 7. At $550\,\mathrm{km}$ the mean value of expected GRBs is 124.19, the standard error of the mean is 0.16 and the standard error of the standard deviation is 0.11. Hence, the mean value is estimated to be between 123.71 and 124.67 at 99.73%. The influence of the uncertainties on the estimated mean and on the dispersion of the results is low, as evident from Fig. 7. The proposed scenario is robust with respect to injection and deployment errors.

Co-alignment on LVLH-optimal directions The scenario with pointing strategy defined as co-alignment on LVLH-optimal directions shows similar statistical results with respect to those discussed for the previous pointing strategy. The robustness with respect to the uncertainties is con-

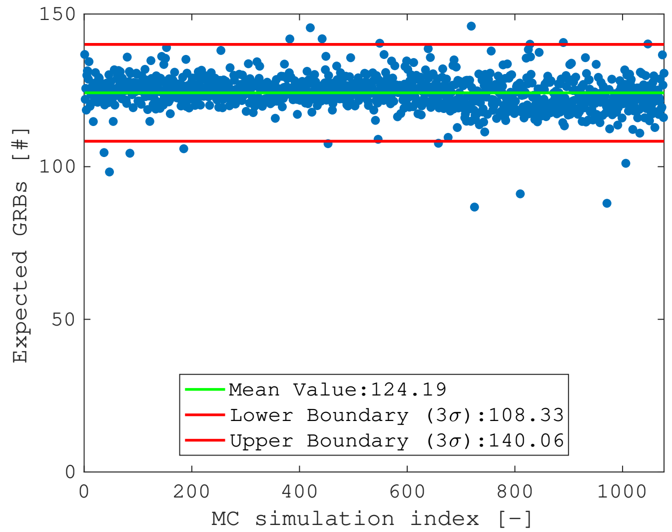

Figure 5: Number of expected GRBs, co-alignment on a LVLH-selected direction: samples

firmed, and the only relevant difference is characterized by a different value for the estimated mean and boundaries of the scientific outcomes. The numerical results are summarized in Tab. 3. Hence, the mean value is estimated to be between 228.98 and 240.62 at 99.73%.

Table 3: Robustness analysis results for the optimized pointing strategy

h [km]	μ [GRB]	3σ [GRB]	SE_μ [GRB]	SE_σ [GRB]
550	234.80	57.03	1.94	1.39

The optimized technique is hereby presented as a showcase on how mission operation can enhance scientific return. These results show the possibility to increase the triangulated number of GRBs, by exploiting at maximum the on-board capabilities of H.E.R.M.E.S.

Non-nominal Mission Scenario

This subsection reports the results for a constellation of 5 satellites. This representative of a scenario in which an early failure of one agent occurs (e.g. one satellite lost during injection or commissioning phase). The following paragraphs present the results for both pointing strategies, namely co-alignment on LVLH-selected directions and optimized pointing.

In the case the adopted pointing strategy is the co-alignment of LVLH-selected directions. The number of triangulated GRBs decreases of nearly 30% with respect to the full 6-sats constellation. Hence, the mean value is estimated to be between 80.98 and 83.06 at 99.73%. Tab. 4 summarizes the numerical results.

Table 4: Robustness analysis results for the 5-sats constellation - LVLH-selected directions.

h [km]	μ [GRB]	3σ [GRB]	SE_μ [GRB]	SE_σ [GRB]
550	82.02	15.08	0.34	0.24

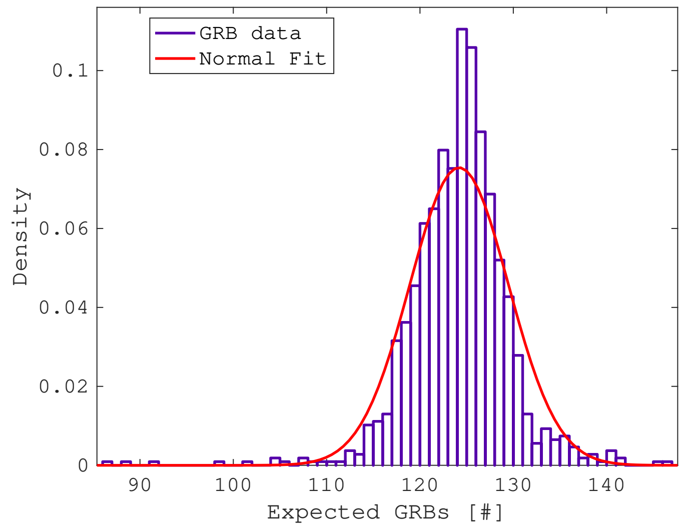

Figure 6: Number of expected GRBs, co-alignment on a LVLH-selected direction: distribution

In alternative, the triangulated GRBs for the optimized pointing strategy using only 5 satellites are more than the yield of the full constellation using the nominal co-alignment pointing approach. In fact, the mean value is estimated to be between 134.29 and 136.67 at $99.73\,\%$. Such result highlights the possibility to enhance the system scientific outcome and the mission robustness by employing optimized pointing strategies. Tab. 5 summarizes the numerical results.

Table 5: Robustness analysis results for the 5-sats constellation - LVLH-optimal directions.

h [km]	μ [GRB]	3σ [GRB]	SE_{μ} [GRB]	SE_{σ} [GRB]
550	135.48	13.29	0.40	0.28

The optimization of the constellation FOV, in terms of elements composing each triplet and in terms of optimal pointing directions, allows to also overcome the loss of an element in the space segment. In fact, when the co-alignment on LVLH-selected direction is used, the possibility to reconfigure the FOV overlap is more limited with respect to the dedicated optimized operations.

POINTING DIRECTION MANEUVERING

The sequence of attitude maneuvers that H.E.R.M.E.S. satellites have to perform in order to change the reference attitude pointing are computed and updated from ground. The schedule of the maneuvers is executed automatically in nominal mode.

Pointing acquisition and pointing direction update (i.e. pointing variation) is performed by means of a tracking pointing on a predefined attitude profile. Hence, the entire maneuver is performed following a pre-defined path.

The attitude profiles for the maneuvers (i.e. pointing attitude acquisition, update pointing direction, telecommunication pointing acquisition, etc.) are computed on-ground and loaded on-board in terms of quaternion profiles. The quaternion profiles are automatically computed on-ground with a dedicated shape-based optimization tool defining the initial attitude state, the final attitude state and

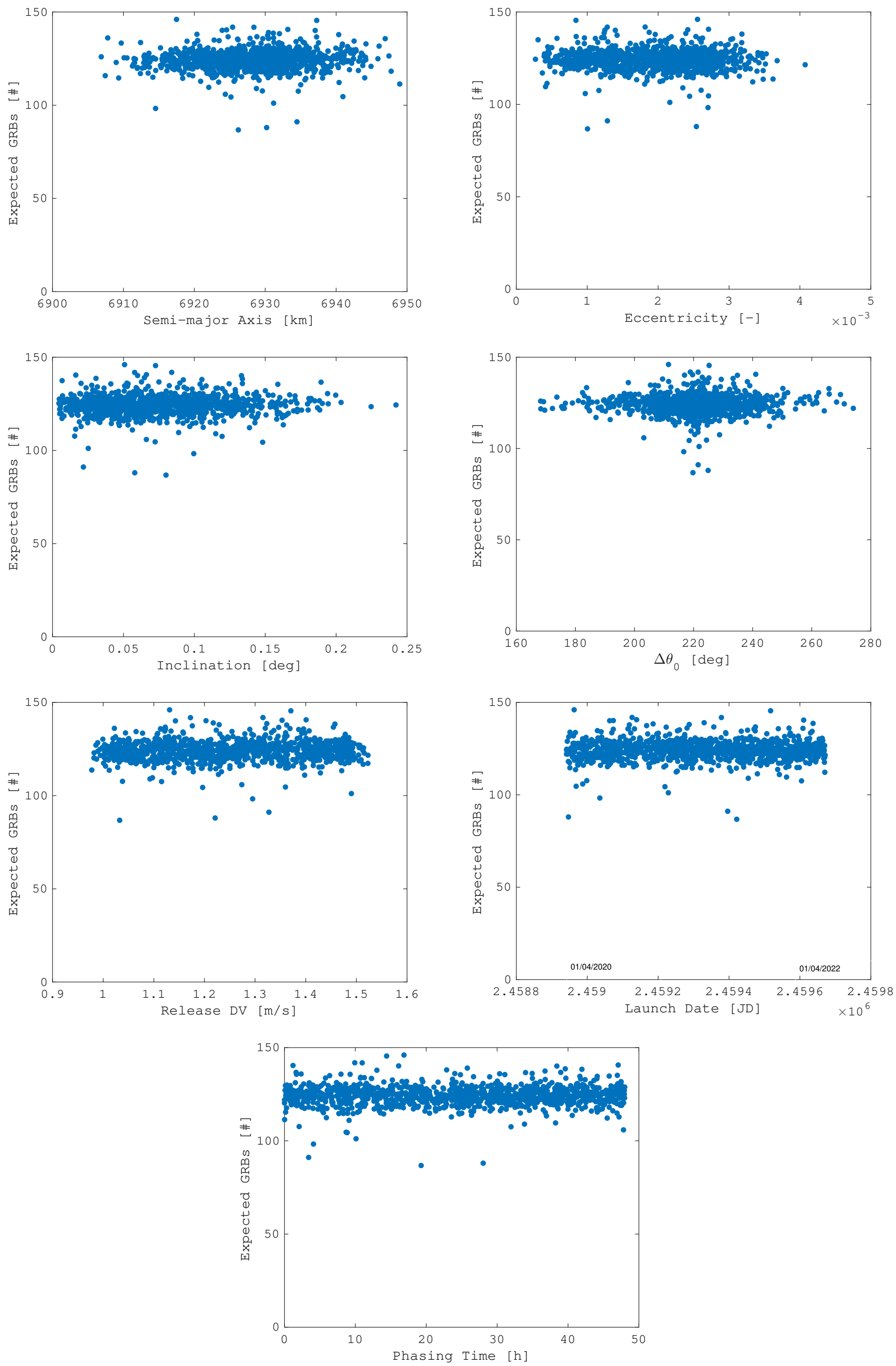

Figure 7: Correlations for co-alignment on a LVLH-selected direction.

the duration of the maneuvers.[3] Moreover, any kind of practical attitude constraints can be included. The attitude trajectory is found minimizing the required control action and, eventually, avoiding particular directions (e.g. the Sun, the Earth) onto the payload line of sight or forcing minimum solar aspect angles with respect to the solar panels. An example of attitude maneuvering trajectory is shown in Fig. 8. In this example, the attitude slew is constrained to maintain the Sun with an angle of 70° with respect to the solar arrays, in order to increase the amount of power produced by the solar panels. This highlights the capabilities of the developed tool. Obviously, whenever no constraint is applied to the attitude maneuvers, the trajectory is found simply minimizing the required control action.

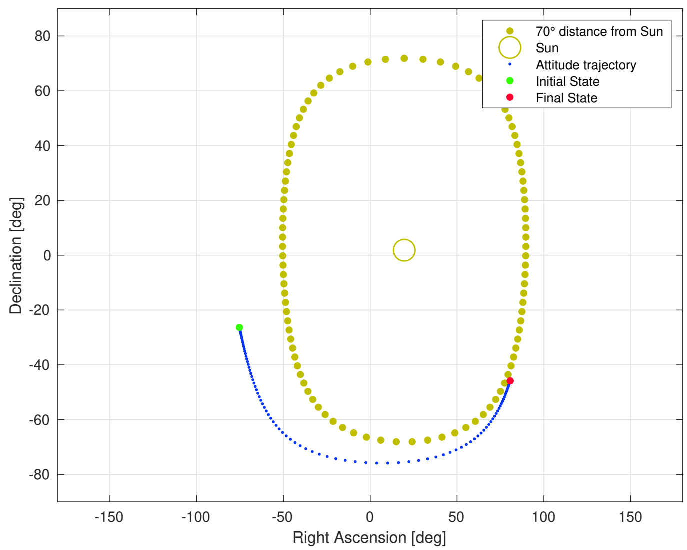

Figure 8: Maneuvering example.

SCIENTIFIC DATA COMMUNICATION

The scientific requirement related with the data volume and the communication performances are extremely challenging for a 3U CubeSat Mission, due to the large volume of scientific data produced and the necessity to fast communicate with the Mission Operations Center (MOC) the presence of an incoming GRB. Among all the requirement related with the scientific data communication, the most important and demanding are listed below:

- **Data Volume:** each satellite shall be able to acquire, store and download $1\ \mathrm{Gbit/day/spacecraft}$ of scientific data.
- **Fast Detection Trigger**: a short message including the most important information on the detected GRB shall be downloaded within $30\ \mathrm{min}$. A smaller time interval of $5\ \mathrm{min}$ is desired.

In order to satisfy the scientific requirements, each H.E.R.M.E.S. satellite is equipped with a directional S-band antenna dedicated to the download of the scientific data, in addition to the classical omni-directional UHF/VHF antenna. With the selected hardware 4 passages per satellite per day are needed to download the scientific data, including a 100% margin. Moreover, in order to be able to communicate the trigger to the MOC at any time, each H.E.R.M.E.S. satellite shall be able to communicate with the commercial IRIDIUM Space Relay Network (SRN).

Ground Segment

The H.E.R.M.E.S. ground segment is composed by two Ground station: Malindi, Kenya, and Katherine, Australia, as in Fig. 9. The geographical coordinates of the GS are reported in Table 6.

Table 6: Ground Stations coordinates

GS	Latitude [deg]	Longitude [deg]
Malindi	-2.995713889	40.19495556
Katherine	-14.3755	132.152

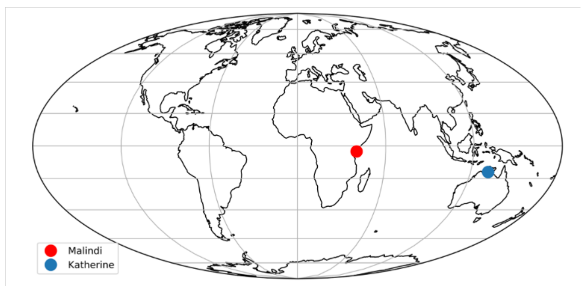

Figure 9: Ground Stations Location.

For the analysis of the communication with ground, the orbits of the six H.E.R.M.E.S. CubeSats have been propagated for one year, using an high fidelity orbital propagator;[4] a margin of $5\,\mathrm{deg}$ on the satellite elevation have been included in order to take into account any possible masking effect. In the H.E.R.M.E.S. mission, the communication mode is one of the most demanding one for the EPS, and the solar panels alone are not sufficient to guarantee enough power, therefore batteries are fundamental to reach the level of power required. In order to reduce the number of cycles of the batteries, the possibility to communicate in eclipse only, at least in the nominal scenario, have been successfully explored.

Communication in eclipse and sunlight In this case, all the passages are considered as possible communication windows; the most important results are reported in Table 7. All the communication windows occurring in one day for one satellite are reported in Figure 10. Table 7 shows the results for the communications only with Malindi, only with Katherine and with both the considered ground stations. The latter case refers to any communication window regardless for the specific ground station, and ca be compared with Figure 10 without distinguish between red and blue line (i.e. the specific ground station is not considered). In this case, a periodicity of a single orbit is evident, with a short revisiting time between the two ground stations (i.e. from Malindi to Katherine) and a longer one for the orbit section from Katherine to Malindi. The combinations in use of the two ground stations results in a long comm-free time, if a communication window on Malindi is followed by one on Katherine, or in a short comm-free time, if the vice-versa occurs. The possibility to have the two communication windows alongside in a single orbit can be considered during the operations, due to the geographical contiguity of the two ground stations along the orbital ground track (i.e. the difference in longitude is $\sim 90°$). In this case, the satellite's antenna is oriented before the passage over Malindi and maintained pointed to ground until the passage over Katherine is concluded. As a result, the ADCS operations could be facilitated, obtaining a longer contact duration with a single maneuver for communication pointing. The analyses report a mean contact duration of $16.73\,\mathrm{min}$ with the two ground station used in a single orbit.

Table 7: Results considering all passages

GS	Malindi	Katherine	Malindi + Katherine
Mean contact duration [min]	10.25	6.48	8.37
Number of passages per year	5146	5146	10292
Revisiting time	102.15	102.15	28.1 or 74.1
Comm-free time	91,9	95,67	17.9 or 67.9
Time with 1 sat in visibility	143.26	107.93	251.19
Time with 2 sat in visibility	32.55	13.51	46.06
Time with 3 sat in visibility	3.79	1.35	5.14

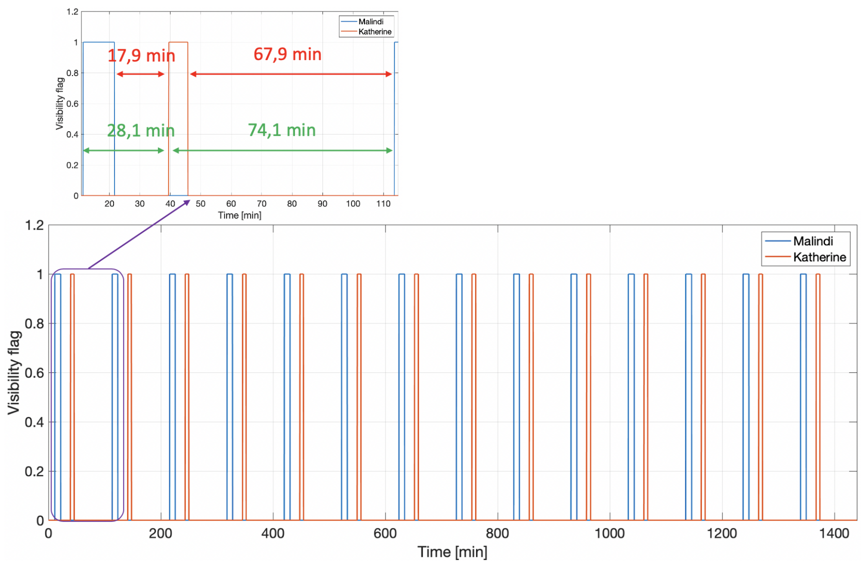

Figure 10: All communication windows in one day.

Communication in eclipse only In this case only the passages in local night-time are considered as possible communication windows; the most important results are reported in Table 8. All the communication windows occurring in one day for one satellite are reported in Figure 11. Table 8 shows the results for the eclipse communications only with Malindi, only with Katherine and with both the considered ground stations. The latter case refers to any communication window regardless for the specific ground station, and ca be compared with Figure 11 without distinguish between red and blue line (i.e. the specific ground station is not considered). In this case, a periodicity of a around $24\,\mathrm{h}$ is evident. In fact, as already said, the two stations are not very distant in longitude and, for a quarter of the orbits in a day, the communication windows occurs with both station in eclipse, resulting in a combination of revisiting time and comm-free time as the one discussed in the previous section. For other two quarters, the two ground stations are alternatively in eclipse and in sunlight (i.e. ground stations in the dawn/dusk regions), and the mixed use of the two locations is not possible. For the remaining quarter, the two ground stations are in sunlight and no communication is possible. Hence, the periodicity of the revisiting and comm-free time shall consider this daily variability. For this reason, in Table 8, two different values of revisiting and comm-free time are reported:

- the first is referred to the case in which two passages in eclipses are consecutive;
- the second is referred to the worst case in which two passages in eclipses are separated by passages in sunlight.

The solution presented in the previous paragraph, which considers a unique passage that includes both the ground stations, is not reported in Table 8 since, as can be seen in Figure 11, due to the not very distant longitude of the two stations, there are only three common passages in eclipse on one day. However, when the two ground stations in eclipse are contacted in one orbit, exploiting only a single slew maneuver, the available mean contact duration is analogous to the corresponding case discussed in the previous section.

Table 8: Results considering only passages in eclipse

GS	Malindi	Katherine	Malindi + Katherine
Mean contact duration [min]	10.25	6.48	8.37
Number of passages per year	1891	1891	3782
Revisiting time eclipse/sunlight	102.15/1013	102.15/1013	from 26,7 to 540
Comm-free time eclipse/sunlight	91.6/1000	95.6/1000	from 18 to 530
Time with 1 sat in visibility	55.11	40.30	95.41
Time with 2 sat in visibility	10.96	4.71	15,67
Time with 3 sat in visibility	1.24	0.45	1,69

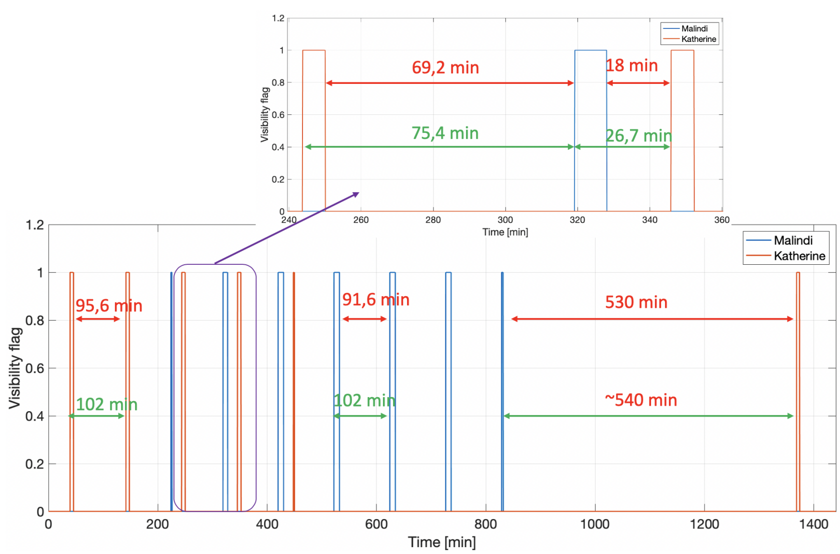

Figure 11: Communication windows in eclipse in one day.

The results discussed in this section prove the possibility to guarantee at least 4 passages per satellite per day. A minimum contact duration of 5 minutes is feasible for all ground stations, guaranteeing the download of the complete data package. The short time of a ground station with more than 1 spacecraft in visibility, as evident in Tab. 7 and 8, allows to maintain a clear separation between the distinct passage of different elements in the constellation.

Space Relay Network

In H.E.R.M.E.S. mission a Space Relay Network is fundamental in order to guarantee the fast trigger to the MOC when a GRB is detected. The most promising SRNs are GlobalStar and Iridium, but for system compatibility Iridium was selected. Iridium is a fleet of LEO communication satellites composed by 66 elements (75 including spare satellites). The satellites are located in 6 SSOs at $780\,\mathrm{km}$ of altitude with same inclination but shifted in LTAN. On each orbit there are 11 satellites shifted in true anomaly. The Iridium satellites are always nadir pointed, with a FOV of $75\,\mathrm{deg}$, while the antenna embarked on H.E.R.M.E.S. satellites has a conservative FOV of $60\,\mathrm{deg}$. Since the attitude of the H.E.R.M.E.S. satellites is not constant in time, as described in , the Iridium coverage analysis is performed with respect to the LVLH reference system instead of the physical axis of the satellites. Nominally, the Iridium antenna is oriented in a body axis close to the LVLH plane containing the t-bar and the h-bar. Figure 12 shows the Iridium coverage for each LVLH axis. In particular for the angular momentum axis there is at least one Iridium satellite in visibility with H.E.R.M.E.S. satellites at any time during the mission, while for the transverse directions there is a $20\,\%$ of time with no coverage. Due to the similar altitude of IRIDIUM and H.E.R.M.E.S. orbits, the worst case happens when the antenna in aligned with the radial direction. However, this condition is rare to occur. Figure 13 shows the probability distribution of the latency between two consecutive communication windows with Iridium satellites. A maximum latency of $\sim 120\,\mathrm{s}$ in the t-h plane is compatible with the nice to have maximum delay of $5\,\mathrm{min}$. The compliance with the required maximum delay of $30\,\mathrm{min}$ is always guaranteed.

SUGGESTIONS FOR POSSIBLE EXTENSIONS

The H.E.R.M.E.S. project is a distributed space asset that would naturally benefit from the enlargement of the satellite constellation. Indeed, the H.E.R.M.E.S. Full Constellation (FC) is under investigation in order increase the scientific performances. The design of the constellation should pursue the following objectives answering the shortcoming inherently present with a single equatorial constellation of six satellites:

- the GRB triangulation is not uniquely determined by satellites orbiting on the same plane. In particular, the hemispheric ambiguity of the GRB localization is resolved inserting different orbital planes
- the full sky coverage, along the mission, is achieved only inserting additional orbital planes. This is due to the intrinsic definition of the projected baseline (i.e. physical baseline projected along triplet mean pointing direction): satellites on equatorial orbits always possess null projected baseline if pointed towards polar regions of the celestial sphere

A proposal for the Full Constellation design is to increase the number of satellites, keeping the number of GRB detected per satellite constant. The rationale is to maintain a limited number of satellites to achieve scientific requirements. This can be implemented by adding satellites on equatorial orbits and at least few satellites on other planes, such as those typical of SSO orbits at the given altitude. The optimization routine executed to determine the preferred pointing directions is a

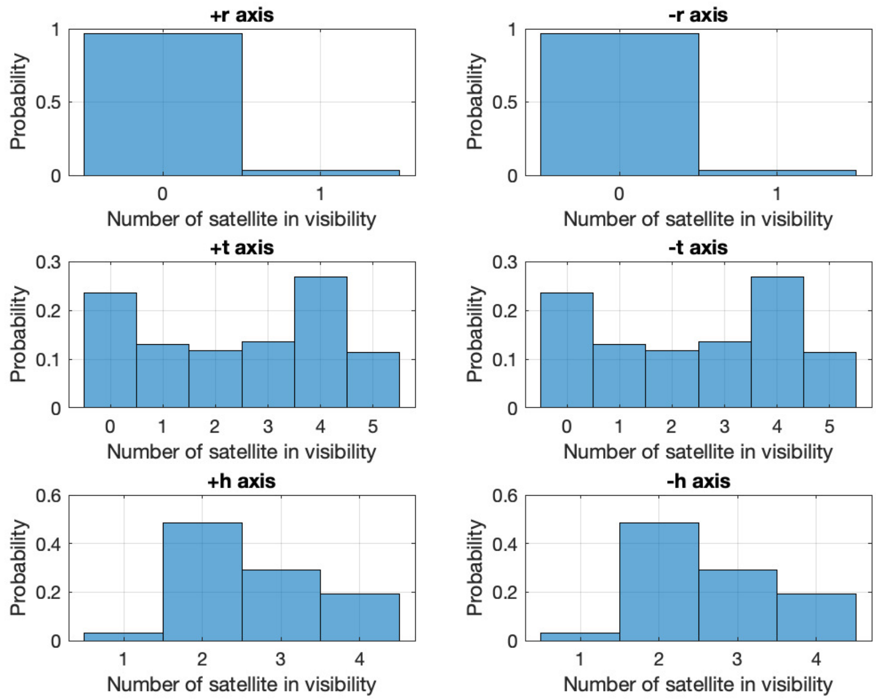

Figure 12: Iridium Coverage

complex task to be executed in the flight dynamics activities. Hence, the Full Constellation should be designed simplifying such operation and keeping a constant attitude throughout the mission, or at least minimizing the required pointing maneuvers.

CONCLUSIONS

The paper presented a comprehensive mission analysis for the H.E.R.M.E.S. constellation, a six-satellite ensemble for the detection and localization of Gamma Ray Bursts to be launched in 2022. The mission analysis features an extensive investigation on injection strategies, natural dynamical behavior and pointing optimization to maximize the scientific yield. In particular, the proposed strategy tackles the mission shortcomings deriving from the employment of propulsion-less platform and limited launches/injection availability. The spring-controlled simultaneous injection has been selected in order to reduce the uncertainties on the resultant relative motion. The imposed motion achieves a drifting relative dynamics, in order to acquire the physical baseline. An optimization routine, based on an in-house sky visibility tool, has been developed to guarantee the fulfillment of the scientific requirement on the number of detected GRBs. Moreover, due to the required quick detection alert, the satellites need to rely on a Space Relay Network to satisfy the time constraint. Hence, the analysis and assessment of ground contact and IRIDIUM link has been presented. The considered ground segment features the ground stations in Malindi and Katherine, which resulted as sufficient for the mission purposes with a mean contact duration of ~ 10 minutes and frequent revisit time. The results for the IRIDIUM contact demonstrated the architecture feasibility, given the number of satellites always visible by H.E.R.M.E.S. constellation (> 4). Specific suggestions

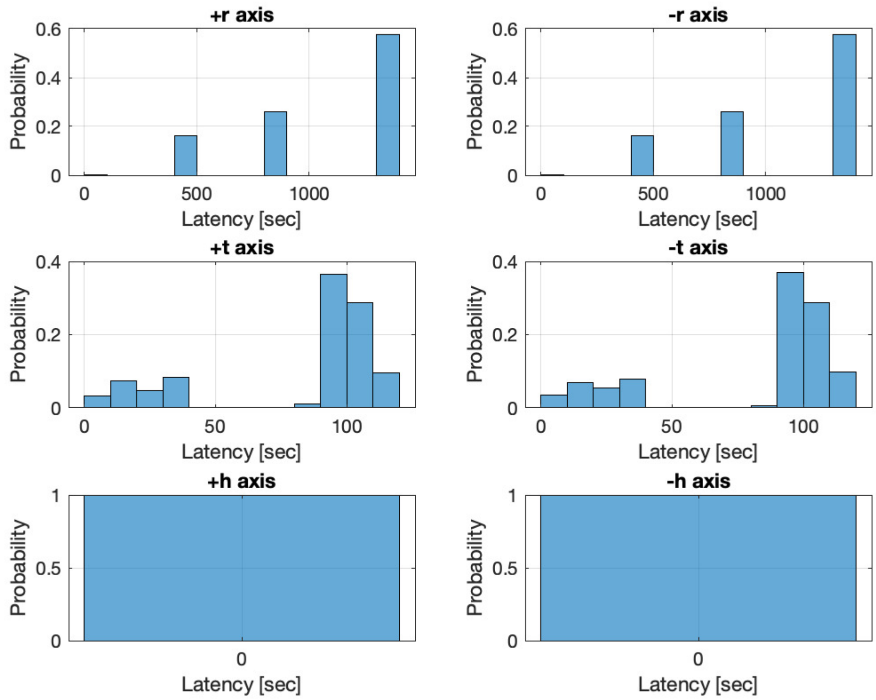

Figure 13: Latency between Iridium contacts

on possible extension of the current design towards a full constellation, within the H.E.R.M.E.S. project, have been presented and discussed. Beside the specific results, the methodology presented in this paper can be easily applied to all the mission entailing a coordinated constellation for sky, or Earth, observation.

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