HENON – Main Challenges of a Space Weather Alerts CubeSat Mission

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Abstract— This paper highlights the main challenges encountered in the phase A and B design of a Space Weather Alerts mission performed by a 12U CubeSat, along with the proposed solutions and mitigation approaches. HENON (HEliospheric pioNeer for sOlar and interplanetary threats defeNce) mission phases A and B have been conducted by a consortium led by Argotec under European Space Agency contracts within the General Support Technology Programme through the financial support of the Italian Space Agency.

HENON is mainly targeted at the provision of alerts upon detection of potentially harmful Space Weather phenomena, namely Solar Energetic Particles and interplanetary perturbations such as High-Speed Streams and Interplanetary Coronal Mass Ejections. HENON will be equipped with state-of-the-art miniaturized scientific payloads to measure energetic particles fluxes, solar wind parameters and interplanetary magnetic field. It will be operated in a Distant Retrograde Orbit (DRO) in the Sun-Earth system, which has never been explored before by any kind of spacecraft. This kind of orbit allows the CubeSat to be favorably placed for a significant period to perform both scientific measurements and alert provision. The transfer to such an orbit is planned to be performed autonomously, employing on-board electric propulsion, which will be another first time ever for a CubeSat in Deep Space.

This ambitious mission brings significant technical challenges. The nature of the mission poses a major challenge, as HENON is supposed to achieve its main objective specifically during Solar Events, which are often a cause for satellites' failure or misbehavior, even for bigger class satellites. For a CubeSat-class spacecraft the challenge is even greater, as the classical mitigation strategies cannot be applied straightforwardly due to the limitations in available resources; hence a different approach based on sector analysis and radiation-driven subsystems placement must be used. To provide its Space Weather alert service, both Sun pointing of the instruments and Earth pointing of the antenna are required simultaneously for about a fourth of the orbit, leading to a trade-off between the ability to acquire relevant data and the ability to communicate alerts to Ground, also considering the long distance from Earth, which already imposes significant challenges for telecommunication. The autonomous transfer to the operative orbit brings supplementary challenges. The electric propulsion imposes a demanding requirement in terms of power generation, especially for a 12U CubeSat. This also implies prolonged thrusting arcs, without communication, with 1 year of transfer time to be added to the 1-year operational lifetime, for a total of at least 2 years in a harsh Deep Space environment where radiation single event effects need to be accounted for and mitigated. Furthermore, the extended mission lifetime and the continued thrusting arcs impose a high level of on-board autonomy, mainly to be implemented at On-board software level, to ease the burden of prolonged and costly Deep Space Ground station operations. HENON is going to be a unique mission, employing breakthrough technologies for a CubeSat in Deep Space. This paper presents the design approach to overcome these challenges and ensure a significant scientific return, reinforcing the path towards CubeSats able to achieve missions previously reserved to bigger-class satellites.

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1. INTRODUCTION

The HEliospheric pioNeer for sOlar and interplanetary threats defeNce (HENON) mission was conceived to address the unanimously recognized and urgent need of making significant advances in the Space Weather field. The mission combines purely scientific research aspects with a service-oriented demonstration of Space Weather Alerts provision.

After a brief description of the mission, this paper focuses on the different challenges that were encountered within the phase A and phase B design, along with the proposed solutions and mitigation approaches.

2. MISSION OBJECTIVES

The needs that prompted to the establishment of the HENON mission have resulted in the following Statement of the Mission:

"To make a leap forward in the space weather effects forecasting horizon and accuracy through in situ observations on unexplored Distant Retrograde Orbit (DRO) in the Sun-Earth System.

To improve knowledge in fundamental plasma process and heliosphere physical mechanisms, demonstrating the reliable use of CubeSat technologies in Deep Space for this kind of applications."

This was the starting points in defining the Mission Objectives which are:

1. demonstrate near real-time in-situ monitoring of the space environment in Deep Space for Space Weather forecasting tools and services.

2. demonstrate the provision of timely near real-time alerts with significantly increased warning times when upstream of the Earth.

3. operate a spacecraft in the unexplored DRO orbits for the first time.

4. demonstrate the reliable use of CubeSat technologies in Deep Space.

HENON will demonstrate the timely provision of alerts upon detection of potentially harmful Space Weather phenomena. Thanks to its payload suite and innovative on-board algorithms, it will be able to detect Solar Energetic Particles (SEP), Galactic Cosmic Rays (GCR) rapid decrease of intensities, i.e., Forbush Decreases (FD), and interplanetary perturbations such as Interplanetary Coronal Mass Ejections (ICMEs) and Solar Wind High Speed Streams (HSS).

Moreover, the spacecraft will perform in-situ measurement of the Deep Space environment with the aim of providing useful data to enhance the currently available Space Weather models by facing with the technological challenge of operating a CubeSat in an unexplored orbit in the Sun-Earth binary System.

2.1. PAYLOADS

To cope with such ambitious Mission Objectives, the entire System was designed to serve the miniaturized Space Weather instrument suite carried on the Platform. Driving much of the mission operations design and spacecraft configuration, the HENON payloads represent the beating heart of the Mission.
Therefore, in the following sections, a technical understanding on payloads functions and performances is provided. Their accommodation within the Platform is shown in Figure 2-1.

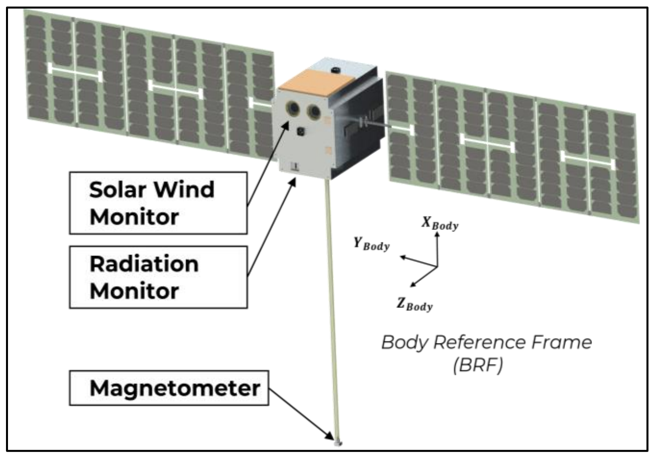

Figure 2-1. HENON Payloads: Solar Wind Monitor, Radiation Monitor and Magnetometer accommodation within the Platform.

2.1.1. Radiation Monitor

To detect electrons and protons over a broad energy range, a compact radiation monitor is embarked on the Platform.

Its main function is to monitor the radiation environment of the mission by measuring energetic protons (from 6 MeV up to more than 100 MeV, with 30% of energy resolution) and electrons (from 0.4 MeV up to 8 MeV, with 20% of energy resolution at energy lower than 1 MeV and 50% for energy greater than 1 MeV). The methods behind the implementation of these functions are the ΔE-E and ΔE-ΔE ones, i.e., the particles' species and their energy are identified by measuring the energy loss signals.
These measurements will be fundamental for the detection of the Solar Energetic Particles (SEPs) and Galactic Cosmic

Rays (GCRs) for the Space Weather Alert Service demonstration as well as to map the particles anisotropy via the spacecraft spinning. To meet the former objective the instrument is continuously pointed sunward.

The key products which guarantee the instrument functions and performances are three Si detectors and one scintillation crystal detector (Gadolinium Aluminum Gallium Garnet, doped with Ce) with photodiode readout. The detector stack is accommodated within the modular structure of the instrument; it consists of two full-size boards for the power supply and digital signal processing, a smaller board attached to the detector unit that hosts amplifiers, and a motherboard that connects all the boards and holds the Platform interfaces. The collimator, with an opening of 40°x20°, allows pitch-angle scanning with an angular resolution of 15° (it corresponds to 24 sectors of 360° and it is mainly achieved by utilizing the spacecraft spin).

An overview of the REPE (Relativistic Electron and Proton Experiment) instrument developed by University of Turku/ASRO is presented in Figure 2-2.

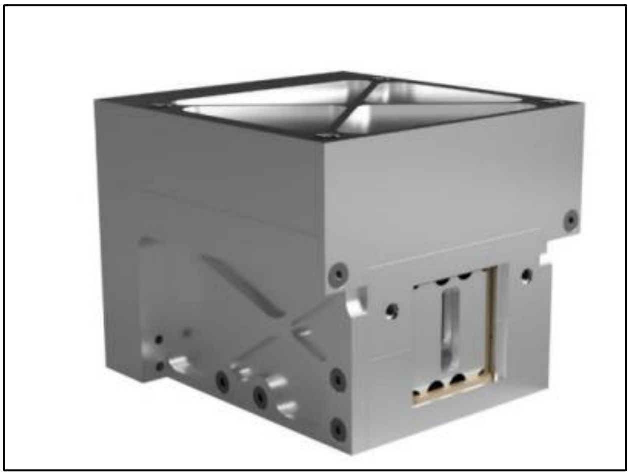

Figure 2-2. Radiation Monitor Instrument (credit: University of Turku/ASRO).

2.1.2. Solar Wind Monitor

To monitor the Solar Wind parameters a Faraday Cup Analyzer (FCA) is selected for the Platform.

It is designed to measure the integrated energy distribution of solar wind ions and will provide the basic moments of the energy distribution in terms of Ion density (from 0.1 cm^{-3} to 200 cm^{-3}), Ion flux (from 0.01 to 50 10^8 cm^{-2} s^{-1}), velocity vector (from 200 km/s to 1000 km/s), and proton temperature (from 0.1 eV to 100 eV) as well as a full 1D velocity distribution of protons and alpha particles.
These data, along with those coming from the magnetometer (see Section 2.1.3), will be of particular importance for the detection of Interplanetary Coronal Mass Ejections (ICMEs) and Solar Wind High Speed Streams (HSS).

The instrument relies on a pair of identical Faraday cup (FC) devices oriented in the solar direction in order to integrate a full solar wind ion flux and to facilitate in-flight calibration. These devices are mounted on top of an electronic box which is made up of three main modules, namely, the Main Board Electronics, the High-Voltage and Control unit and the Preamplifier and Measurement unit. To detect the phenomena needed to meet the mission objectives, the instrument needs to be pointed toward the Sun.

The FCA instrument developed by Charles Technical University is showed in Figure 2-3.

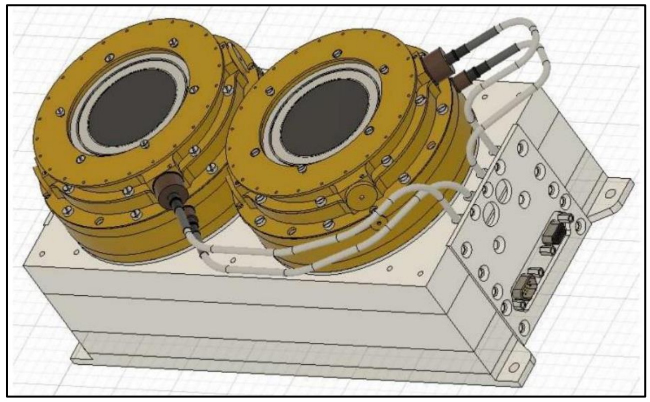

Figure 2-3. Faraday Cup Analyzer (FCA) instrument (credit: Charles Technical University).

2.1.3. Magnetometer

The measurement of the Interplanetary Magnetic Field (IMF) will be entrusted to a compact and low resource magnetometer.

The Magnetometer makes precise measurements of the magnetic field (in the field range on ± 60000 nT with a digital resolution of 7 pT and a sample rate from 0.1Hz up to 10 Hz) which will be merged with the ones coming from the Solar Wind Monitor for the detection of geo-effective ICME and HSS. The knowledge of the relative orientation of the Platform and the instrument itself is sufficient to reconstruct the three components of the magnetic field vector, thereby, the instrument does not impose any particular pointing constraint to the HENON Spacecraft.

These measurements are guaranteed by two Anisotropic Magneto-Resistive (AMR) vector sensors placed within the two functional blocks of the instrument:

- The sensor head, comprising the hybrid magnetic sensing element (outboard sensor, on top of a deployable boom system),

- The electronic card, which houses the magnetometer control electronics.

The two functional blocks are connected via ultra-low mass harness.

An overview of the instrument developed by Imperial College London (ICL) is presented in Figure 2-4.

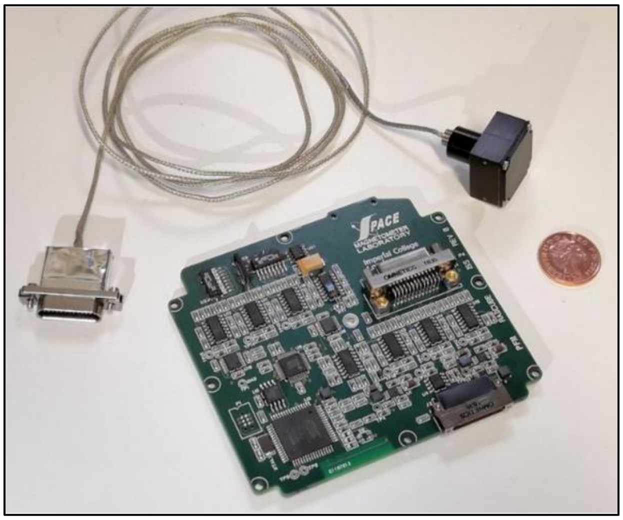

Figure 2-4. Magnetometer Instrument: outboard sensor, harness, and inboard electronic card (credit: ICL).

3. MISSION OVERVIEW

The HENON Mission Concept of Operation is divided into five main phases: Launch and Early Orbit Phase (LEOP), Transfer, Payload Commissioning, Nominal Operational and Disposal.

The LEOP (1) is defined as what happens from being released (about 2 hours after launch on a trajectory bound for the Sun-Earth L2 Lagrange point) up to and including Platform and main propulsion system commissioning.

The main goals of the phase are to autonomously power up and boot up the Platform, to reduce tumbling rates given by the release, to deploy the solar panels for battery recharge and to wait for the first contact with Ground. These activities will allow the Flight Control Team to take control of the satellite by performing the first communication and checking out all the subsystems as part of the Platform commissioning. Particular attention needs to be given to the propulsion subsystem, which needs to be fully commissioned within 17 days of the release to start the first thrusting arc, to avoid any negative impact on the baseline trajectory.
The start of the first thrusting arc will bring HENON in its second mission phase.

The Transfer Phase (2) is going to be started within 17 days of the release; it represents one of the most critical and challenging scenarios among all mission phases, thereby, it is described in more detail in Section 3.2.

The main goal of this 1-year transfer is to reach the target operational Distant Retrograde Orbit (DRO) by means of the on-board electric propulsion system and weekly communications windows needed to maintain the spacecraft.

After the target orbit insertion, the Payloads Commissioning (3) phase will be used for the checkout and calibration of the scientific instruments, in order to enable HENON entering its Nominal Operational Phase (4, see Section 3.1) during which the Spacecraft will monitor the radiation environment demonstrating the near real-time alert service to the Earth and acquiring first-ever measured in-situ data until being disposed (5).

3.1. OPERATIVE ORBIT

3.1.1. Overview

As discussed in the Section 2, the mission challenge to fly for the first time ever a Distant Retrograde Orbit comes from the need to make a leap forward in the Space Weather forecasting horizon. While this kind of orbit has been already employed by several spacecraft in the Earth-Moon System, they are completely unexplored in the Sun-Earth one.
This is a heliocentric orbit with the same orbital period as the Earth, resulting in a quasi-elliptical retrograde orbit as seen from the Earth in the Sun-Earth rotating frame.
An effective and useful production of Space Weather predictions requires that these predictions are made with sufficient advance time to set up protection and mitigation measures by the stakeholders. Clearly, the largest the advance time, the more effective the mitigation measures can be.
Spacecraft in L1 can provide early measurements of space weather perturbations, but, given that solar wind perturbations travel with a speed of 400 km/s at least, the advance time is of the order of one hour or less. In a recent review paper, Vourlidas et al., [1], recommended deploying a spacecraft sunward of the Earth at 0.3 AU from the Earth, with the aim to have warnings with an advance time of the order of 24 hours. As a step in this direction, HENON, being displayed at about 0.1-0.08 AU in front of the Earth, will be able to detect events with an advance time of the order of 4-10 hours (depending on the speed of the perturbations). This represents a quantum improvement with respect to the prediction which can be given by spacecraft in L1.
Also, the stability of the DRO allows for virtually no Station Keeping maneuvers, which are a must for spacecraft orbiting L1. On the other hand, due to the orbital dynamics the spacecraft cannot be stationary along the Sun-Earth line. In fact, HENON is intended as an in-orbit demonstrator and a precursor of a constellation of at least four properly phased spacecraft in the DRO, to provide continuous coverage on the sunward side.
In the DRO, HENON will be conveniently located in between the Sun and the Earth for about one fourth (timewise) of the orbit, at a distance far greater than L1 (Figure 3-1).

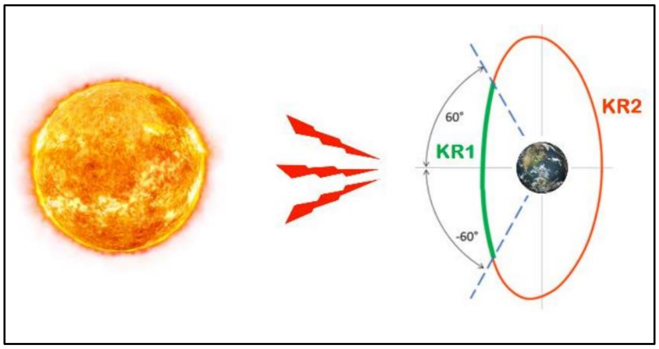

Figure 3-1. HENON DRO, highlighted the key regions of the mission.

3.1.2. Baseline target DRO

Early in the design phase, a systematic approach was implemented in establishing a suitable target orbit among the ones of the DRO family. This process played a crucial role by finding the optimal solution between the scientific/ service return and technological feasibility. An overview of the process and its results are reported in the following.

First, the key-driver requirements were defined to find out feasible options. They are: to guarantee an alert service to the Earth with reasonable advance with respect to occurred events as well as with respect to the current services in L1, to require less than 3km/s of ΔV to reach the DRO, to guarantee an achievable data rate sufficient to provide both the alert service and the spacecraft health data, and to guarantee a two-way emergency communication link for any satellite attitude.

The identified candidate orbits were traded-off via the decision-making method of the Analytical Hierarchy Process, [2], on the basis of five parameters, i.e., required transfer time, achievable data rate, propellant mass and distance from Earth. Moreover, to overcome the human lack of consistency in judging (due to emotional, experience, etc.), the evaluation was based on the judgment of several engineers making up the HENON mission design team.

To properly estimate some of the trade-off parameters, the transfer phase had to be studied in conjunction with the target orbit selection. The transfer phase is described in Section 3.2.

The results led to the identification of a preliminary baseline orbit which, at the end of a refinement during the mission/system preliminary design phase (namely, Phase B), show that the orbit which HENON would be going to fly shall feature the following orbital parameters in the Sun-centric MJ2000ECL - T0 (UTC): 14-Jan-2023 at 11:33:33 - (Figure 3-2):

- SMA ~1.0 AU
- e ~ 0.067
- i~0.36°
- RAAN~273
- AOP ~36
- TA~167°

Therefore:

- DRO minimum distance from Earth of about 12.3E6 km
- DRO maximum distance from Earth of about 24.52E6 km
- DRO minimum distance from the Sun of about 0.93 AU
- DRO maximum distance from the Sun of about 1.07

3.1.3. Operations

The operational orbit is divided into two key regions, namely Key Region 1 (KR1) and Key Region 2 (KR2).
The KR1 includes the longitude range [-60, +60]° with respect to the Earth-Sun vector. This region is the one that will allow HENON to demonstrate the near-real time alert service for the Earth. Indeed, thanks to the payloads pointing sunward, the spacecraft will monitor the particle radiation environment and the geoeffective interplanetary perturbations.
The payloads' measurements will be evaluated on board to compute indexes relative to the different monitored threats (SEPs, FD, ICME, HSS). Upon overcoming predefined and programmable thresholds, an alert message is generated and stored on-board for downlink at the earliest opportunity (see Section 5).

The KR2, defined as the part of the orbit complementary to the KR1, is therefore exploited to better characterise the radiation environment through the high-resolution measurements of the three payloads and to download data to Earth. In addition, an Anisotropy mode has been defined to make the spacecraft perform a spin around its X axis at a rate of 1.5°/s once certain space weather events are detected (SEPs and FDs). This allows a measurement of the anisotropy of the radiation field, further improving the mission's scientific results.

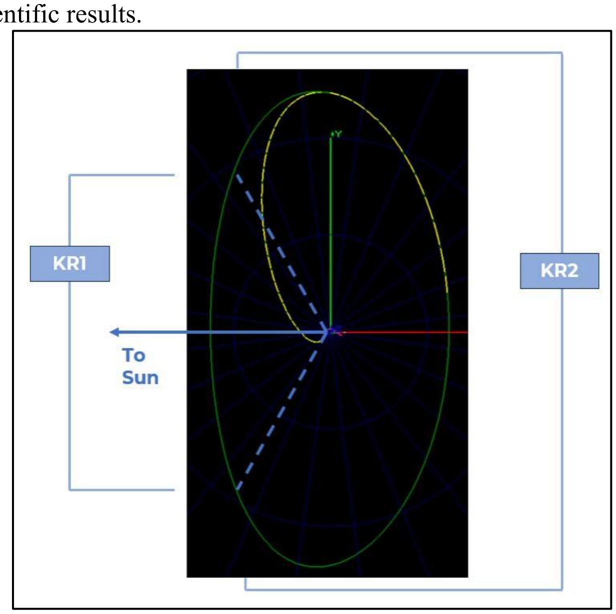

Figure 3-2. Overall mission trajectory: transfer (yellow) and DRO (Green).

3.2. Transfer phase

Such a challenging operational orbit in the Sun-Earth System brings several difficulties in reaching it: the autonomous 1-year transfer phase of HENON around our star represents a disruptive concept in the CubeSat assets, posing another first-time ever in the mission. With the aim to demonstrate the reliability of these small platforms in harsh environment of Deep Space, HENON will try to overcome those technical hurdles previously reserved for larger satellite classes.

The design challenges associated with the present scenario mainly impacted the trade-off among the different orbital transfer options, suitable on-board propulsion subsystems, power gathering strategies, radiation hardeness assurance, operational cost, and schedule. As a matter of fact, a large number of such challenges could be overcome accommodating the Platform inside a Carrier Vehicle. Indeed, it would bring several advantages such as not having to keep the CubeSat switched on during the time it takes to reach the operational orbit and shielding from radiation due to its accommodation inside the Carrier itself, highly de-risking the mission. However, in the current state of the art, no Carrier Vehicles are available that are qualified for Deep Space and able to meet the high delta-V requirements for the DRO transfer.
A market survey was conducted to understand the possibility of a dedicated development or adaptation of existing Carriers, however this was eventually deemed incompatible with the project budget. An on-board propulsion solution was then considered, accepting the drawback of longer operational time and associated cost of flight operations.

Therefore, the key-design solutions implemented to tackle the 1-year autonomous transfer phase are presented in the following.

To reach the operational orbit, the System embarks a high-total impulse and low-mass Electric PROPpulsion subsystem (EPROP, see Figure 3-3) which relies on the Radiofrequency Ion Thrust technology to perform the transfer with a low-thrust strategy (on-board chemical propulsion was ruled out for available volume and mass limitations).
This technology guarantees the best performances in terms of total impulse with, however, the drawback of the relatively large power demand for this size of platform. To satisfy the power request of the propulsion system and to not constrain the thrust profile of mission, the System embarks a Solar Array Drive Unit (SADU) developed by IMT which will keep the Solar Arrays pointing to the Sun to maximize the gathered power during the transfer. The SADU, although being a relatively standard technology, certainly represents a mission enabling component, as it allows to achieve the maximum power generation to feed the propulsion system, maximizing the generated thrust and the thrusting time, while also avoiding imposing constraints on the thrust direction.

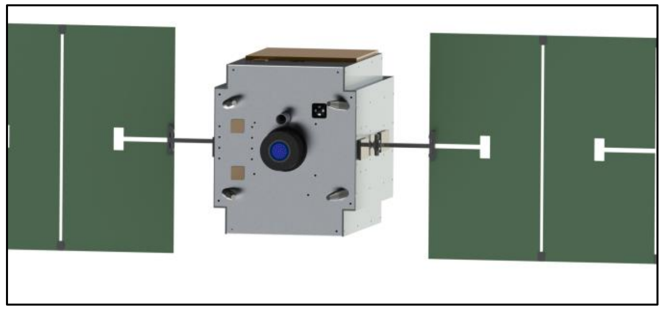

Figure 3-3. EPROP General Overview.

Along with the high-power request of the electric propulsion, a low-thrust strategy, fixing the total impulse of the mission, implies also high transfer times. The current baseline foresees a trajectory of about 385 days. In this context, it is crucial to guarantee satellite commandability and observability, investigating the issue with analyses such as:

- Trade-offs on the duty-cycle, in order to identify efficient and safe timing for alternating thrust and communications. In general terms, periodical sessions of communication and ranging activities are necessary during the entire mission, in particular to perform orbital determination, wheels off loading and to keep track of the SC orbital motion. To balance the need for limiting the transfer time and Platform management, the current HENON baseline foresees six days of continuous thrust and one day of no thrust to allow 8 hours of communication.
  Simultaneous operation of the propulsion system and TT&C system is ruled out by the available power. Details on the power challenge of the mission is described in Section 7.
- Envision, when possible, features which add autonomy for the time frames in which the satellite is performing maneuvers without visibility of the ground stations. The capabilities as well as the reliability of these autonomous controls shall be deeply verified and together with the impact of non-nominal behaviors.
- Involve, early in the design phase, the Ground Station provider in order to define a baseline availability for the communication window. This will help avoiding unfeasible operations design and timeline, guaranteeing a more robust end-to-end System.
- Optimize the on-board antenna positioning to avoid a weak communication and ensure reachability of the spacecraft should the ground segment need to take control of it outside the scheduled communication windows. This challenge is detailed described in Section 5.

Moreover, the harsh operational environment is a main driver for the System availability and survivability requirements. Hence, it has to be properly characterized and taken under

consideration as one of the main criteria for subsystems selection and trade-off. The approach to this topic is described in Section 6.
One of the first mitigation approaches was to limit the transfer time, hence limiting the exposure to the radiation environment.

The design of the transfer trajectory took into account all the previous considerations, with the additional assumption that the spacecraft will be launched as a secondary payload of a larger mission to the Sun-Earth L_2, and that the separation from the launcher occurs a few hours after launch. Typical launch conditions of this kind put the spacecraft in a high-elliptical Earth orbit (apogee ~10^6km). A launch towards L1 was also considered at the very early stages of the trajectory design, but it was later discarded due to the lack of available launch opportunities in the considered timeframe, although the system design remains compatible with L1 launch opportunities should they become available.
Other options, such as a transfer from LEO or GTO were explored and discarded due to the excessive propellant mass, transfer time requirements, total ionizing dose exposure and overall complexity. Similarly, a dedicated launch to obtain the best possible initial conditions for the transfer was discarded due to budget limitations.

In order to simulate a realistic initial condition for the HENON transfer trajectory, and being unknown, at this stage of the Project, the initial condition of the identified piggyback launch opportunity, the SPICE kernel of the transfer trajectory of the NASA James Webb Space Telescope (JWST) mission to L2 were exploited up to two hours from the launch (C3 ~ -0.7km^2/s^2).

Since the core part of the mission starts once the spacecraft reaches KR1, it was also desirable to approach the DRO on the righthand (anti-sunward) side, as seen in Figure 3-2, so to have enough margin to properly calibrate the payload while not waiting for too much time before reaching KR1.

This led to the exploitation of a heteroclinic connection between L2 and L1, so to achieve the desired point of insertion in the target orbit. This connection is shown in Figure 3-4 as a green line from separation to the first start of thrusting, as a magenta line from first thrust up to escape from the Earth sphere of influence. The full transfer trajectory is shown as a yellow line in Figure 3-1.

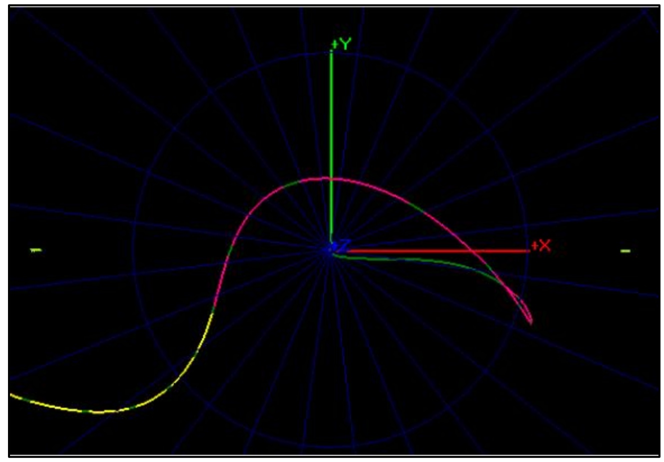

Figure 3-4. Geocentric phase of transfer after separation from launcher, in the Sun-Earth rotating frame.

This also led to a beneficial side effect, as the spacecraft will perform its transfer in the area closer to the Sun, hence being able to harvest as much power as possible, increasing the Propulsion System thrust level and consequently reducing the total transfer time.

4. Platform Overview

This section provides an overview of the baseline Platform as the outcome of the phase B design, with the aim of giving adequate context for the understanding of the mission challenges.

4.1. TT&C

The TT&C system is responsible for making communication between the satellite and the ground stations possible. This system consists of an X-band radio developed by IMT (enabling both uplink and downlink communication and ranging) and 6 antennas, namely 2 TX low gain single-patch, 3 RX low gain single-patch, and one TX high gain array. Their particular arrangement on the Platform (see section 5 for details) guarantees that the spacecraft can be reached in any attitude and at any point of the trajectory. The radio guarantees 15W of RF output power, necessary to establish a link even under contingency conditions.
It is worth mentioning that the link budgets and visibility analysis were performed assuming the usage of the ESTRACK network (15m and 35m stations), however the usage of commercial assets are currently under consideration in an attempt to increase the availability especially in critical mission phases.

4.2. OBC&DH

The Fermi On-Board Computer & Data Handling (OBC&DH) system manufactured by Argotec constitutes the operational core of the spacecraft. Indeed, its responsibility is the control of the various subsystems and data handling through the On-Board Software (OSW). Through the TT&C system the OBC is responsible for sending data to Earth. Its architecture is based on two main components: CPU and FPGA. They both feature a rad-hard design to provide protection against destructive and non-destructive SEEs. The CPU provides the interfaces that ensure complete control of the satellite, while the high-speed FPGA constitutes an extension of the CPU by providing additional interfaces and allowing hardware acceleration, mass memory control, and advanced payload management functions. Particularly important at software level is the Fault Detection, Isolation & Recovery (FDIR) system, which is critical for handling failures and to prevent catastrophic issues.

4.3. Structure & Mechanisms

The structure performs the task of supporting the hardware constituting the Platform and keeping the various subsystems in the correct position. A custom 12 U structure will be developed by Argotec for HENON to provide this functionality. The spacecraft will also integrate the following mechanisms, supplied by different providers:

- Boom system: to deploy and hold the Magnetometer outboard sensor outside of the spacecraft (Astronika).
- Hold Down Release Mechanism (HDRM): to deploy the solar panels (IMT).
- Thruster Pointing Mechanism (TPM): to orient the thrust vector in the desired direction and avoid buildup of parasitic momentum (AVS).
- Solar Array Drive Unit (SADU): to enable the tracking of the Sun when needed (IMT).

4.4. ADCS

The ADCS system performs the task of determining and controlling the satellite attitude. The system selected for the HENON Platform consists of 4 reaction wheels, an electronic control board, a star tracker, a set of gyroscopes, and a set of fine sun sensors. It already demonstrated robust performance in Deep Space and it is compliant with the momentum storage and pointing accuracy requirements.

4.5. EPS

The EPS subsystem is in charge of generating, converting, storing and distributing power to the various subsystems. It consists of two wings of solar panels (5 panels per wing) attached to a SADU (Solar Array Drive Unit) mechanism to continuously point the Sun, a PCDU (Power Conditioning & Distribution Unit) and a battery to store power and release it when generation from the panels is not available or when the power margin is negative. The PCDU from Argotec features a rad-hard design and galvanically isolated primary and secondary lines. It also provides protection against electrical faults (e.g., overvoltage, overcurrent) and its main task is to distribute the generated power to the subsystems.

4.6. Propulsion System

The HENON propulsion system consists of an EPROP integrated with a Reaction Control Subsystems (RCS). The former performs the task of providing the necessary thrust to carry out the transfer, while the latter is used for momentum management of spacecraft (i.e. wheel off-loading). The EPROP is composed by a Radio-Frequency Ion thruster, a propellant-less neutralizer, a Thruster Pointing Mechanism (to orient the thrust vector with respect to the spacecraft CoM), Power Processing Units (PPUs), a Flow Control Unit (FCU) and a Propellant Storage and Management Unit (PSMU), to manage and store the Xe propellant at high pressure. With less than 5kg of mass, it can guarantee more than 3500s of Specific impulse and 1.7 mN of thrust at 110W input power.

The EPROP shares the same Xenon tank with the integrated cold-gas RCS system, which consists of 4 thrusters that can provide a torque level in the range of 5-50 mNm.

This propulsion system represents an enabling technology for HENON mission, and it is currently under development within a separate dedicated ESA GSTP R&D activity by a consortium led by Mars Space Ltd.

4.7. Platform summary

An overview of the Platform deployed configuration is show in Figure 4-1 , while Table 4-1 summarize its main features.

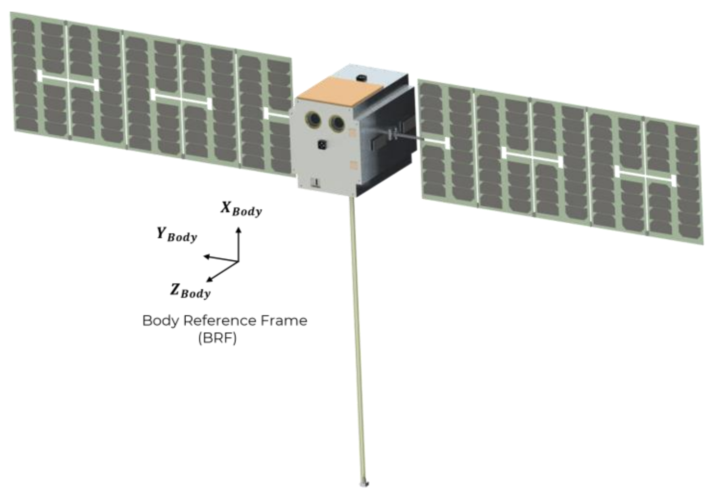

Figure 4-1. HENON Rendering, Deployed Configuration.

Table 4-1. HENON Main Features.

Feature	Value
Mass	29 kg
Volume	12U+
Downlink Band	X-Band up to 512 kbps
Lifetime	> 2 years
Power generation	> 200 W
Payloads	Radiation Monitor, Faraday Cups, Magnetometer

5. ALERT PROVISION AND TELECOMMUNICATIONS

The generation of on-board alerts and subsequent dispatch to Earth is the core of the mission, posing further technical challenges to be met with judicious design choices.
As mentioned in Section 3.1.3, a severity index for each monitored type of threat is computed on board, on the basis of the data collected by the payloads.
As a representative example, the index relative to the SEPs events will be computed every minute processing measurements provided by the Radiation Monitor. A positive index indicates the detection of a SEP event and its actual value, based on the NOAA Space Weather Scales for SEPs events, [3], will also provide a preliminary evaluation of the severity.
Similar indexes will be computed for the other types of monitored events.

Whenever at least one event is detected, an alert message is generated and stored on a dedicated Packet Store on board. After the first detection an alert packet will be generated every 3 hours (interval programmable in flight), to provide a first estimation of the event severity trend.
As a way to check the proper functioning of the on-board algorithms, an alert packet will also be generated every 12 hours in case no event is detected. This will act as a dummy packet containing all null indexes.
The alert message is defined to keep its size as small as possible to keep the required data volume low; it contains just the indexes and the detection timestamp.

The alert Packet Store will be downlinked to Ground, as a high priority, every communication session (due to cost constraints the ground station coverage is not continuous in the KR1 region for this demonstrator mission). The current CONOPS foresees Ground to command the downlink of such Packet Store as one of the first actions at the beginning of a communication session.
However, alert messages are also routed directly to Ground for direct downlink if a link is established.

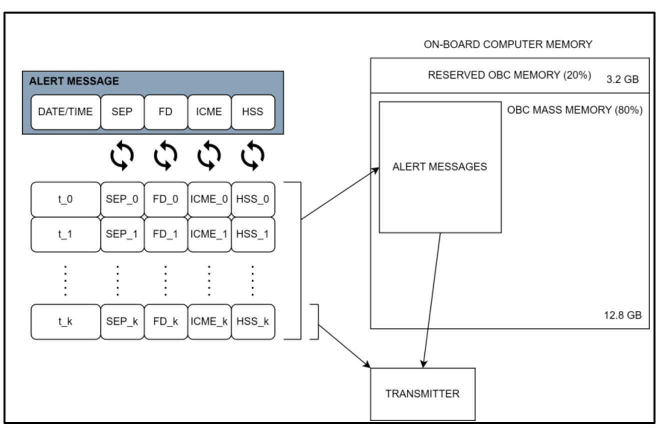

Figure 5-1. Architecture of Alert Message during Communication Windows.

Figure 5-1 represents the Alert generation and transmission system.
The design challenge posed by this particular service is to be able to have the payloads pointing towards the Sun continuously while simultaneously communicating to Earth, so as to ensure near real time alert service. This requirement heavily impacted the trade-offs at communication subsystem and spacecraft configuration level. Indeed, these two operations are apparently in conflict with each other, as they may require different pointing and thus a different Platform attitude, thus disrupting continuous payload operations. The key design solution foresees optimized arrangement of the antennas, placing patch arrays on the opposite face of the payload (-Z), so that the geometric characteristics of the antenna's cone of view guarantee continuous ground visibility while the satellite is located in KR1 (Figure 5-2). This ensures fine pointing of the Sun by the payloads combined with the possibility of establishing communication with Earth.

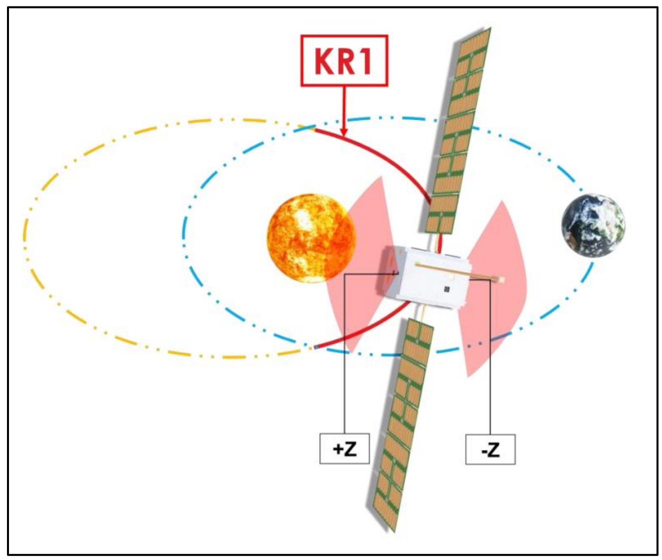

Figure 5-2. Representation of HENON in KR1 for Alert Service to the Earth while Pointing Sunward (eccentricities accentuated).

A further technical difficulty of real-time measurements of solar storm events is achieving a level of availability such that there are no instrument or Platform reboots at such a high frequency that the service is affected. Indeed, the spacecraft shall be able to withstand and operate during the harsh conditions of Solar Storms. A better discussion of the radiation environment and the related strategies adopted is described in Section 6.

6. RADIATION ENVIRONMENT

Unique is the environment through which HENON shall be able to demonstrate its Objectives. The Platform aims to operate during those Solar Events which would pose huge malfunctions even in bigger class satellite. Due to the CubeSat class of the Mission and related volume and mass constraints, the challenges requested to the Platform cannot be overcome by exploiting mission-critical subsystem redundancy. This section illustrates the HENON environment in the various mission phases, its impacts on the Platform and the envisaged mitigation strategies.

During the LEOP, the most critical phase is the passage through the Van Allen belts. In fact, although the passage lasts only a few hours, its impacts cannot be neglected.
Being the HENON spacecraft expected to be released at ca. 20,000 km altitude, no high energy trapped protons are expected to be faced with during the pass since the energetic trapped protons population (above 10 MeV) is confined to altitudes below 20,000 km (lower energy protons population are present, up to the geosynchronous altitudes, but with protons energy below 1 MeV). These considerations are confirmed by the analysis results for both non-destructive and destructive events. However, more than 17% of the Total Ionizing Dose (TID) is given by the 5 days of the LEOP (in face of less than the 0.7% of the total mission time). This contribution to the TID of the whole mission is due to the environment itself which the spacecraft will face; indeed, differently from the other mission phases, during the LEOP, beside the solar protons' contribution (which are the only major source of dose from transfer to nominal operations), even trapped protons, electrons and Bremsstrahlung are present.
At the end of the Transfer Phase, the accumulated radiation doses are more than half of the ones of the entire mission.
During the operational phase, the main hazards are represented by destructive and non-destructive SEEs which will impact survivability and availability of the System. This is mainly because the spacecraft is required to operate during radiation critical solar events. The need to take measurements and send alerts to the ground during these events precludes the possibility of using Safe Mode or shutting it down to avoid radiation-related criticalities. This poses a great challenge in terms of radiation hardening.

As a general approach, to tackle this technological challenge, a radiation analysis of the mission is performed according to the worst-case models. Four main radiation sources are identified:

1. Trapped particles (electrons and protons) fluxes over the spacecraft trajectory.
2. Solar particle mission fluences predicted for the total mission duration (long-term effect).
3. Solar particle peak fluxes predicted for individual events (short-term effects) considering the worst 5 minutes peak flux averaged over 5 minutes (observed in October 1989).
4. GCRs fluxes predicted considering the 1996 Solar minima.

To derive the environmental specifications through which drive the subsystems selection and/or delta-design, the main impact on the Platform (TID, TNID, soft and hard SEE, Solar Cells radiation damage and Charging) are taken into account considering 100% of margin.

For all the above reasons, radiation assurance is one of the most important criteria when selecting subsystems. In addition to a selection criterion, it is also a driver for the configuration and, where possible, the architecture of the System. Detailed FMEA, availability and sectoring analyses are the tools used to identify mitigation actions such as:

- Radiation Driven Placement
- Robust FDIR for non-destructive event management, at both software and hardware level
- Anti latch-up devices
- Spot shielding
- Delta design of subsystems or components not compliant with the environmental specification.
- Radiation test campaign to test those mission critical components with no traceability.

It must be emphasized that the proposed solutions have impacts on masses, volumes and mission costs and must therefore be carefully evaluated.

7. POWER GENERATION

The Platform is particularly power demanding. This is due to the presence of two high power consumption subsystems such as the electric propulsion system and the X-band radio. The propulsion system is crucial during the transfer phase, as is the X-band radio, which will also be used during near real time service in KR1 and data downloading in KR2. The total power required to ensure sufficient power with margins in all operating modes is approximately 200 W, which is higher than what typical solar panels for 12U CubeSats can provide. The adopted solution is focused on custom solar panels, composed of two wings consisting of five panels each. Quadruple-junction solar cells were considered to ensure the highest efficiency. In addition, the panel assembly is equipped with a SADU (Solar Array Drive Unit) to optimize solar panels exposure to sunlight, thus maximizing power generation during all mission phases. The SADU will be equipped with rad-hard components to protect the subsystem from the harsh radiation environment that the Platform will face in Deep Space. Managing such a significant input power load and its conditioning and distribution to the various subsystems also requires an improvement of the current PCDU technology available to CubeSats. Argotec Volta PCDU, which successfully flew on LICIACube [4,5] and ArgoMoon missions, and represents the state of the art for Deep Space CubeSat technology, will be subject to a dedicated rework to match the demanding requirements of HENON.

8. CONCLUSIONS

In conclusion, at the current stage, the HENON mission is presented with a complex, ambitious and innovative design. The various choices made are aimed at satisfying the high-performance requirements, while at the same time respecting the mass and volume constraints imposed by the Platform's form factor. The final result is a Platform that pushes the boundaries of the current state of the art for 12U CubeSats in terms of System performances, mass, autonomy and lifetime required. The technical challenges are many, among them providing the satellite with adequate radiation protection but also the challenge associated to the autonomous transfer phase. As next main steps, the HENON flatsat model will be developed, adequate qualification campaigns for the most critical components will be assessed, the System and subsystem analyses will be consolidated (mechanical, thermal, power, etc.) and further improvements will be studied with the aim of achieving a robust and optimized design for the Critical Design Review. Flight readiness is planned for the late 2026 timeframe, and various launch opportunities to Sun-Earth L2 in that timeframe are currently under consideration.

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BIOGRAPHY

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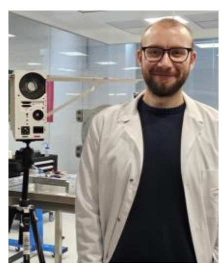

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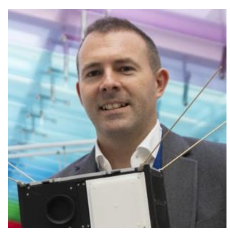
Simone Landi *(M) (OrCID 0000-0002-1322-8712) PhD in Astrophysics and Space Science, associate professor at the University of Firenze in Astronomy and Astrophysics, male, has more than twenty years of experience in the field of plasma physics of the solar corona and solar wind, working on developing and using innovative numerical techniques and also on data-analysis and reduction. He is author of more than 80 peer reviewed papers with more than 2000 citations (H-index 28) collaborating with researcher worldwide. He is actually involved in national programs on plasma astrophysics and in the scientific team for the METIS instrument on board of solar Orbiter (ASI- INAF). He is also PI and collaborator of several projects for numerical computation both national (CINECA-ISCRA) and European (PRACE) for using Tier- 0 machines. It is also involved in the board for the local High Performing Computing infrastructure of the University of Florence, and it is the supervisor of the local Cluster of the Physics Department. It is actually involved in the scientific support for the ASI-ESA CubeSat mission HENON in collaboration with INAF staff.*



Roger Walker *received his Ph.D. from the University of Southampton in 2000. He is currently the Head of the CubeSat Systems Unit in the Projects Office of the Systems Department, Directorate of Technology, Engineering & Quality at the European Space Agency. He has been pioneering the use of CubeSats in the Agency since 2008, first as hands-on education tools for universities, then in the last decade leading technology in-orbit demonstration missions aimed at advancing their state-of-art and diversifying their mission utility for various applications.*