

Article

Design, Implementation, and Operation of a Small Satellite Mission to Explore the Space Weather Effects in LEO

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Abstract: Ten-Koh is a 23.5 kg, low-cost satellite developed to conduct space environment effects research in low-Earth orbit (LEO). Ten-Koh was developed primarily by students of the Kyushu Institute of Technology (Kyutech) and launched on 29 October 2018 on-board HII-A rocket F40, as a piggyback payload of JAXA's Greenhouse gas Observing Satellite (GOSAT-2). The satellite carries a double Langmuir probe, CMOS-based particle detectors and a Liulin spectrometer as main payloads. This paper reviews the design of the mission, specifies the exact hardware used, and outlines the implementation and operation phases of the project. This work is intended as a reference that other aspiring satellite developers may use to increase their chances of success. Such a reference is expected to be particularly useful to other university teams, which will likely face the same challenges as the Ten-Koh team at Kyutech. Various on-orbit failures of the satellite are also discussed here in order to help avoid them in future small spacecraft. Applicability of small satellites to conduct space-weather research is also illustrated on the Ten-Koh example, which carried out simultaneous measurements with JAXA's ARASE satellite.

Keywords: space environment; plasma; radiation; single events; particle detectors; Langmuir probe; magnetometer; rapid spacecraft development; small-satellite; Ten-Koh

1. Introduction

Space radiation affects satellites by introducing anomalies such as single event effects (SEE), component degradation due to ionizing radiation dose, and surface and internal charging. Understanding the radiation environment is, therefore, important in order to design satellites that can withstand the possible anomalies. Space radiation sources include galactic cosmic rays (GCR), solar energetic particle (SEP) events, high energy particles trapped in the Earth's magnetic field and the continuous radiation background. In addition, the low-Earth orbit (LEO) region where most satellites reside is subject to unknown mechanisms that provide it with unpredictable energy variability in the spectra of particles [1–3]. This variability is particularly poorly understood for electrons, which can appear with energies higher than expected.

The consequences of the presence of high-energy electrons, protons and ions for spacecraft developers vary depending on each mission design, epoch and class. The spacecraft design should account for the effects of ionizing radiation, as well as charging and discharging effects on satellite surfaces. For manned missions, mission duration [4,5] and the associated life support systems need to be tuned to account for this unpredictability of particle populations.

In recent years, different missions have been launched to explore the near-Earth region in order to provide direct measurements of charged particles, plasma, and the magnetosphere. Missions such as the Van Allen Probes (RBSP) [6,7], THEMIS [8], MMS [9] from NASA, JAXA's ERG (ARASE) satellite [10], Proba-2 [11] and Swarm [12] from ESA have been launched into specific orbits to study space radiation around the Earth [13]. In addition to the mentioned missions with launch masses of hundreds of kilograms, dedicated small satellites offer an important complement to the measurements because they enable a wider, more comprehensive view of the space environment thanks to their reduced development time and cost. This class of satellites leverages commercial off-the-shelf (COTS) components to bring cost and time savings at the expense of an increased risk of failure. The small-satellite Ten-Koh has been developed to demonstrate the feasibility of providing space environment measurements with such low-cost platforms, as well as to provide readily usable data.

This paper provides a comprehensive overview of the Ten-Koh mission design, presents its preliminary results and discusses the applicability of small, low-cost satellites to conduct space weather research. The specific objectives of the mission are described next, followed by a summary of the satellite platform and payloads. The main challenges in the development of the satellite are then summarized and the adopted solutions to them described. Even though the mission has been successful, which is demonstrated by briefly presenting data obtained in-orbit, it has suffered a number of failures at component, assembly and subsystem level. These are described in detail to help improve the design and implementation of future small satellites that take advantage of the COTS components.

2. Materials and Methods: Ten-Koh Mission Description

2.1. Ten-Koh Project Objectives

The Ten-Koh satellite mission described in this paper has the following primary objectives:

1. To characterize the plasma environment around a spinning spacecraft.
2. To detect MeV-range electrons in LEO and investigate the space environment in the presence of an extreme low solar activity.
3. To investigate the change of physical properties of LATS (Lightweight Ablator series for Transfer vehicle) and CFRP (Carbon Fiber Reinforced Polymer) material samples exposed to the space environment.

Involving students in satellite development, manufacturing, testing and operations is an important part of their curriculum and enhances their education. Providing this involvement has been set as a secondary mission objective of Ten-Koh.

Ten-Koh also provided a flight opportunity for two technology demonstration payloads, which constituted another secondary objective of the mission. These payloads are the Thermal Switch developed at Sigma Space Systems, which flight-proves a novel design of a switchable passive thermal switch [14], and the Ultracapacitor Experiment that aims to quantify the performance of an ultracapacitor as a satellite energy storage device. These two experiments are not discussed in detail in this paper because they are not directly related to particle physics and they do not address the primary mission objectives. However, their location within the Ten-Koh architecture is presented for the sake of completeness.

From the mentioned primary objectives, the novelty of the Ten-Koh satellite derives from observations of space radiation and ionosphere environment at the end of the 24th Solar cycle, with an increase in GCR flux in higher latitudes. This is associated with Primary Objectives 1 and 2. As for Primary Objective 3, resin composites are expected to deteriorate when exposed to the in-orbit conditions, with the deterioration rate dependent upon the current state of the LEO environment. Satellite observations of this environment combined with measurements of the degradation of composite materials make the Ten-Koh spacecraft unique, especially in the class of small satellites.

This paper focuses on the space radiation and its direct effects on electronics, not material degradation. Thus, the focus is placed on Primary Objectives 1 and 2 while others, notably Primary Objective 3, will be discussed in dedicated publications due to their specific nature. However, the accommodation of the instruments required by Primary Objective 3 is described here for the sake of completeness.

2.2. Satellite Platform

Ten-Koh was developed over a period of 16 months mainly at Kyushu Institute of Technology (Kyutech), while some payload instruments were designed and developed in parallel in Australia, Bulgaria, and the USA. Ten-Koh is based on the platform of a previous small deep-space probe Shinen-2 [15], developed and launched by Kyutech in 2014. The main structure of Ten-Koh is composed of a CFRP composite shell with a rigid internal load-bearing structure made of aluminum alloy (Al 6061-T6).

The satellite platform was based on several components and architecture topologies used by Shinen-2. However, the design and integration of most of the platform subsystems has been completely redefined in order to suit the new mission objectives. The main goal when adapting the Shinen-2 platform architecture for the purpose of Ten-Koh was to reuse as many of the heritage electronic components as possible in order to reduce the risk of component failure and thus increase system reliability. The development time was also shortened by reusing the same structure, which assured both successful environmental testing and removed the need for devoting a significant effort to design a new structure. Figure 1 shows the Ten-Koh satellite in its in-orbit configuration.

The satellite platform is formed of the following subsystems:

- OBC—on-board computer and data handling subsystem;
- COMM—communication subsystem;
- EPS— electrical power subsystem; and
- ADS— attitude determination subsystem.

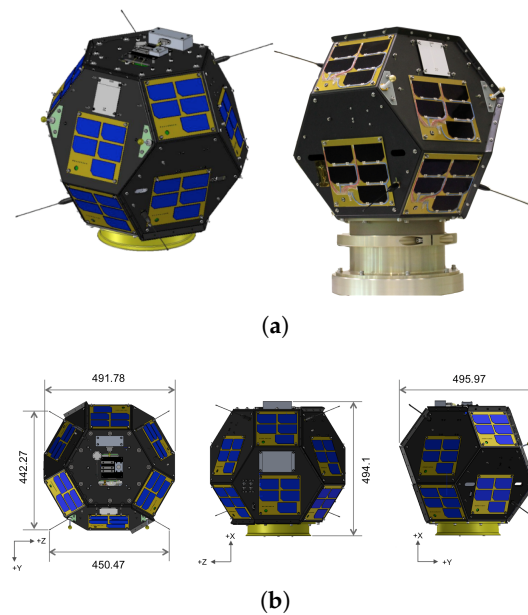


Figure 1. Ten-Koh satellite flight model configuration once in orbit. The black external structure is made of CFRP. (a) Computer-aided design rendering on the left, flight model photograph on the right. (b) The envelope allowed by the launch vehicle was 500 mm × 500 mm × 500 mm.

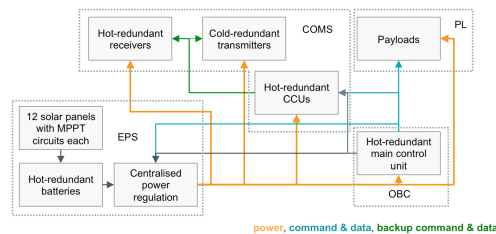
Due to the short time allowed for the development and limited resources, attitude control was not included in the satellite and the use of any mechanisms was discarded, even though they were being considered until the preliminary design review (PDR). A passive attitude control based on permanent magnets was not possible due to the influence on the CPD and magnetometer readings. Figure 2 shows the block diagram of the satellite platform subsystems and the main interfaces between them. Each of the platform subsystems is described below.

2.2.1. OBC

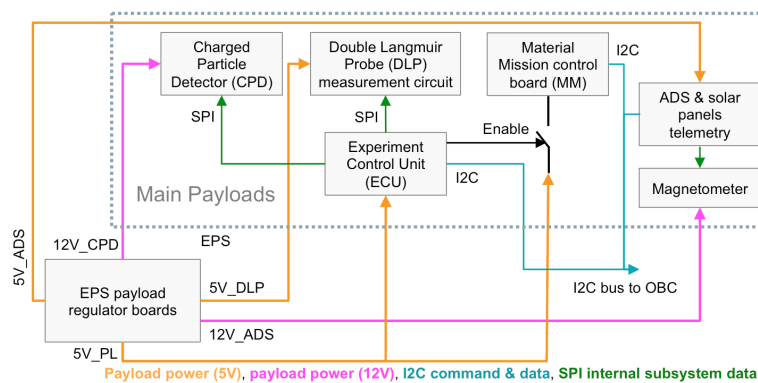
The on-board computer and data-handling subsystem is based on a distributed control architecture, whereby each subsystem is governed by one PIC16F877 microcontroller. This PIC was chosen for Ten-Koh due to its Shinen-2 legacy, which increased the confidence that the controller would function in-orbit. Using the same microcontroller in all subsystems allowed complete re-use of the same circuits or even PCB layouts, which drastically reduced the required development time. Software was also re-used to the maximum extent possible by implementing generic libraries that could be used across many subsystems. In this way, only the top-level application software layer had to be developed for every subsystem microcontroller once the libraries were in-place.

The OBC PIC or main controller unit (MCU) is the master of the satellite-wide I2C data bus, on which all the subsystem microcontrollers are slaves. During the testing of the I2C master-slave communication architecture with several PIC microcontrollers, we found that when any of the slaves was powered off, the I2C communication stopped. The cause of this issue was a drop in the I2C bus voltage, which reduced from the nominal 5 V TTL (transistor–transistor logic) level to around 2 V. By including isolation between the I2C devices that are powered from different EPS power lines, the I2C data communication bus could continue to operate even if any of the slaves was powered off. The isolation was implemented with the ADUM1250 IC, which relies on the same magnetic isolation technology as the ADUM14xx family that has been proposed for small-satellite data handling application [16]. Moreover, because of the isolation, the I2C bus should remain usable even in the case of failure of one or more of the microcontrollers. Failure of an isolation IC itself could, in theory, disable the entire bus. However, due to the use of a flight-proven isolation technology, such risk was minimized. Another SPI data bus is implemented as backup to enable satellite control in case of I2C

bus failures. Due to the limited number of general purpose input/output pins (GPIO) available on the OBC PIC, only the vital subsystems are connected to this backup SPI bus, namely EPS and COMM. In case the I2C bus failed, the satellite would still be controllable from ground and operators would take actions to try and recover full functionality of the spacecraft, e.g., by resetting specific subsystems or the entire satellite. SPI isolation between all the PICs is implemented with HCPL-0631 optocouplers.



(a) Ten-Koh system architecture.



(b) Ten-Koh payload architecture including: experiment control unit (ECU), material mission (MM), charged particle detector (CPD), and double Langmuir probe (DLP).

Figure 2. Block diagrams of the Ten-Koh system as well as its payloads. Dashed lines show the limit of each subsystem and the solid lines indicate the type of interface. Ten-Koh satellite uses I2C as the main data bus (blue), and SPI (gray) as a backup data bus and direct interface with sensors inside every subsystem. The 5 V power lines are shown in yellow and the 12 V lines in magenta. The ADS subsystem was included inside the payload subsystems (PL) for convenience in the physical location inside the satellite, hence it appears in the bottom block diagram.

The OBC PIC implements the main satellite state-machine software, which reacts to ground commands and changing conditions with pre-programmed actions. Other subsystem PICs are only responsible for gathering telemetry from their subsystems and executing OBC commands, with the exception of the EPS and COMM microcontrollers. These two can either activate battery heaters or reset the entire satellite, if such a command is received from the ground.

The OBC PIC constantly polls all the currently active subsystem microcontrollers in a sequence to monitor their status. If any of the subsystems reports a change of state, e.g., a failure, a received command uplink in case of COMM, or completion of a given task, the OBC PIC responds accordingly. On-board time, kept by a DS1340 real-time clock (RTC), is also monitored by the OBC such that it can execute time-tagged commands sent by the ground control.

Analog-to-digital conversion (ADC) on board is performed either by the PIC16F877 in-built ADC, or when more signals need to be sampled, with an eight-channel, 12-bit AD7927 ADC that uses the ADR421 voltage reference. The only exception to this are specific payload instruments, which require more accurate conversion. Data are stored on SD cards, one per subsystem. After reception of an appropriate telecommand, the OBC will transfer the data from the specified SD card into a buffer, which will then be downlinked to the ground. In case 5 V logic level of the PICs needs to be shifted

to 3.3 V, as required, e.g., by the SD card or the RTC, TXB0104PWR and TXS0102DCTR level-shifters are used.

2.2.2. COMMS

Telecommunications subsystem includes two hot-redundant uplink chains and two cold-redundant downlink chains, with one uplink and one downlink chain governed by one microcontroller (communications control unit, CCU). The amateur frequency transceivers with flight heritage were supplied by a Japanese manufacturer (<http://www.nishimusen.co.jp>). These are almost completely self-contained RF circuits and only require an external modem, which was implemented with the FX614 IC.

Both COMM PICs constantly monitor their respective receivers and notify the OBC in case a valid uplink command has been received. The OBC then decides which downlink chain to use in order to send an acknowledgement to the ground-station. Both receivers operate at different frequencies such that the ground control can choose which uplink chain will receive the uplink, and each command specifies which downlink chain the OBC should use. This solution allows flexible in-flight debugging of the system.

2.2.3. ADS

The attitude determination subsystem philosophy is outlined in Figure 3. Raw measurements from coarse sun-sensors, gyroscopes and magnetometer are processed on-ground in order to estimate the attitude. Because attitude control has not been implemented, attitude determination on-board was not required, which further simplified the on-board software.

The 12 coarse sun-sensors were implemented using S4349 photodiodes sampled with the AD7927 ADC, with one diode placed on each solar panel of the satellite. Triple modular hot-redundant A3G4250d gyroscopes were used to provide attitude-rate measurements, while the magnetic field is sensed by the HMC2003 three-axis magnetometer and two 16-bit ADCs. All the ADS sensors are sequentially sampled over a SPI data bus by one PIC microcontroller in order to ensure that all the readings are synchronized, which is required for attitude determination. A detailed description of the ADS subsystem is included in [17].

While the spacecraft is in view of the ground station, the ADS sensor data can be downlinked in real-time and combined with downlink signal measurements done by the ground station. This enables higher attitude determination accuracy during ground station passes.

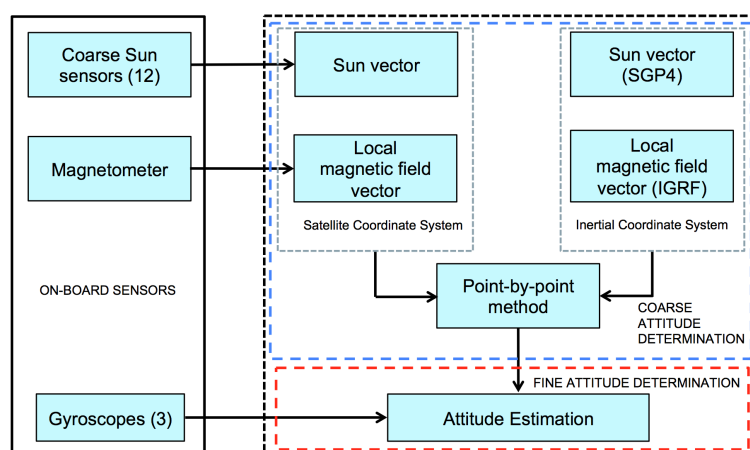


Figure 3. Attitude determination philosophy for the Ten-Koh satellite. The magnetometer instrument is used for space environment purposes as well as for ADS.

2.2.4. EPS

The main power source of Ten-Koh consists of 12 solar panels composed of five triple-junction solar cells each, with two SPV1040 MPPT (maximum power point tracking) circuits per solar panel: one for managing two cells and one for managing the remaining three cells. All 12 solar panels are connected in parallel in an unregulated power bus that is used to charge two hot-redundant batteries. Each of the two batteries is composed of four individual Li-ion cells connected in parallel. The complete outputs of both batteries are also connected in parallel in order to form a battery-regulated bus.

The battery-regulated power bus is an input into the LT1370 voltage regulators, which generate 5 and 12 V regulated buses. These buses then feed over-current protection (OCP) circuits implemented with LTC4361 and LT4363 ICs, which in turn deliver power at regulated voltages to the different subsystems. All subsystems were designed to use these voltages to reduce the complexity of the entire system. In certain cases, however, 3.3 V levels are required. These are obtained from the 5 V lines with LD1117 linear regulators within the subsystems.

The different power lines per subsystem are:

- Two independent 5 V lines for the payloads—one for the experiment control unit, ECU and one for the double Langmuir probe, DLP.
- One 12 V line for the charged particle detector (CPD) instrument.
- One 12 V line for the magnetometer instrument.
- Two 5 V lines, one for each of the redundant communications subsystems.
- One 5 V line for the attitude determination subsystem (ADS) and all solar panels telemetry acquisition circuits.
- One 5 V line for the entire EPS subsystem.
- One 5 V line for the OBC subsystem.

Each power line can be controlled (switched on and off) independently from the EPS controller PIC. The default state of the power lines is encoded in hardware to make sure that the OBC and COMM subsystems are on even in case of failure of the EPS controller. This PIC also gathers voltage, current and temperature telemetries from the entire subsystem.

The total power needed to run all the platform subsystems (OBC, EPS, and COMMS) is 3.3 W. When the payloads are operated, this value increases to at most 9.5 W, which is the case of running the platform plus the CPD and DLP (worst case allowable for payload operation). Depending on the experiment being conducted, the power consumption can be reduced to:

- Material mission (MM): 4.8 W.
- Charged Particle Detector (CPD): 8.8 W.
- Double Langmuir Probe (DLP): 5.5 W.
- Attitude Determination Subsystem (ADS): 5.5 W.
- Thermal Switch (TSW): 5.5 W.
- Ultracapacitor (UCP): 5 W.

From the power generation perspective, the minimum available power is 11 W, which leaves sufficient margin to recharge the battery during sunlit parts of the orbit. A detailed description of the solar panels designed for Ten-Koh is shown in [18].

2.3. Payload

The main payload consists of five individual instruments, the main characteristics of which are summarized in Table 1. One of the main payloads is the Charged Particle Detector (CPD), which was developed by Prairie View A&M University, TX, USA and the Space Research and Technology Institute—Bulgarian Academy of Sciences, Bulgaria. This system consists of eight CMOS

(Complementary Metal Oxide Semiconductor) detectors mounted on five faces of a cube, and the Bulgarian Liulin-type detector [19] mounted on the top of the complete assembly. The CPD instrument contributes to all four primary mission objectives.

Another payload of Ten-Koh is a set of Double Langmuir Probes (DLP). This experiment is intended to study the plasma and the sheath formation around the satellite as it is spinning along its orbit. Satellite charging, which can occur at surface and internal level, is an important subject of study because it is related to satellite anomalies and is directly impacted by the space weather conditions. Satellite charging depends on the interaction of the spacecraft materials, their thickness, and charged particles energy. Fluxes of electrons ranging from 10 to 100 keV can produce surface charging, while electrons with energy bigger than 100 keV can produce internal charging (deep dielectric charging) [20]. The plasma flow around a satellite in LEO exhibits a behavior similar to the flow of a neutral gas: there is a region of compression on the side where the plasma flow impinges upon the moving spacecraft, and a wake region behind the body of the satellite (see Figure 4a). For satellites that have no attitude control, the modeling of the plasma interaction with the satellite body becomes a complex task. Therefore, in-situ measurements provided by the described experiment would be a valuable source of information useful for the validation of the plasma models.

The DLP system is composed of two spherical, 10 μm gold-plated electrodes, as shown in Figure 5a, which are placed outside of the spacecraft structure. The electrodes are mounted on an isolating glass fiber reinforced polymer (GFRP) plate to avoid any electrical contact with the CFRP structure, thus keep the electrodes floating with respect to the spacecraft ground. The shafts supporting the spheres are coated with alumina to ensure that only the spheres attract plasma particles, thus making the interpretation of results easier. The DLP control circuit, housed inside the spacecraft, uses a 16-bit digital-to-analog converter (DAC) to generate the DC biasing voltage. This allows the sweep duration to be tailored such that the plasma in different locations around the spinning spacecraft can be characterized, as shown in Figures 4a,b and 5a,b. The current flowing between the probes through the plasma and the biasing voltage are measured with a AD7927 12-bit ADC. The measurement circuit is isolated from the common ground of the spacecraft, which enables the plasma parameters to be measured.

The purpose of the material mission (MM) is to expose three different samples of new material for space applications developed by Okuyama laboratory of Kyutech made of carbon fiber (CF) and PEEK resin:

- Sample 1: CF/PEEK with no coating.
- Sample 2: CF/PEEK with a special coating (silsesquioxane) to protect against atomic oxygen.
- Sample 3: CF/PEEK with a special coating (yttrium oxide) to protect against UV.

The system measures different parameters, including temperature of and strain inside the samples in order to quantify the changes in the coefficient of thermal expansion (CTE) of the new materials. By measuring the changes in CTE, internal micromechanical modifications in the chemical composition of the samples caused by the effects of space environment can be discerned. Testing in this way allows assessment of material survivability and reliability for use in space. The temperatures are sensed with AD590 temperature transducers sampled by the AD7927 ADC. The ADS1220, a 24-bit ADC, and COTS strain-gauges are used to measure strain in two orthogonal directions inside the samples.

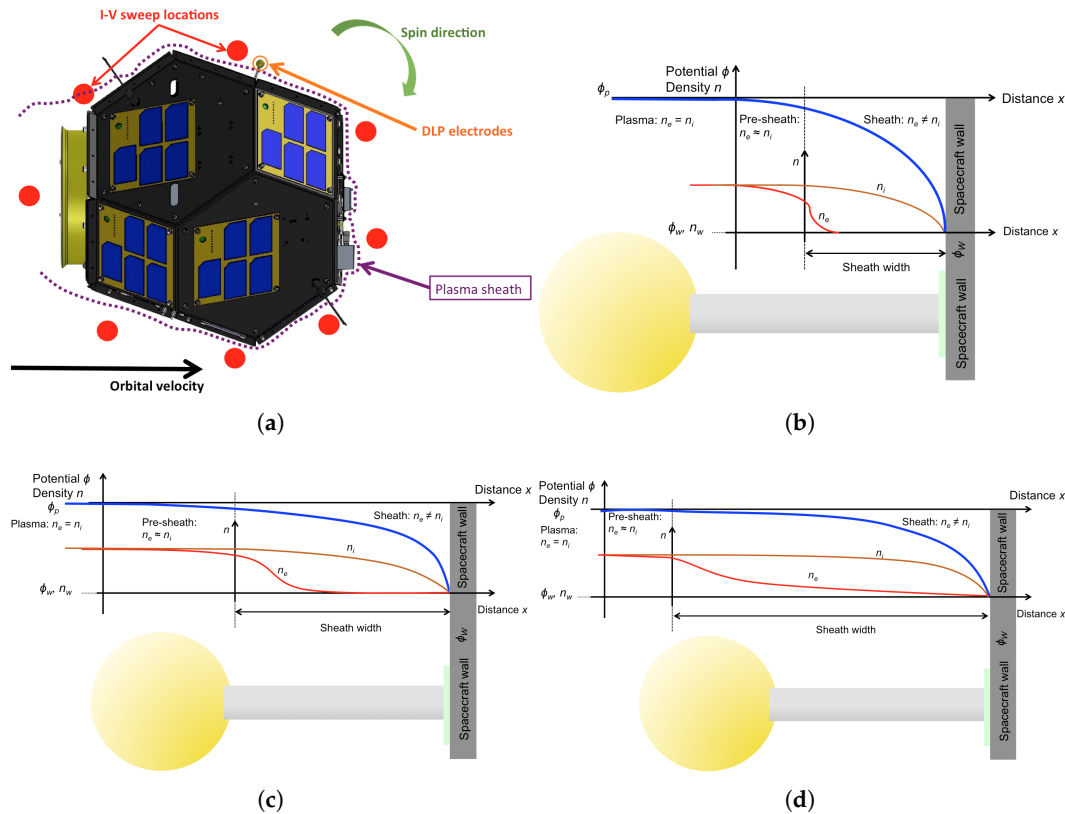


Figure 4. Representation of the DLP measuring system: (a) I–V (current–voltage) sweep locations around a spinning spacecraft; and (b–d) 1-D model representing the electrodes inside and outside of the plasma sheath.

The payloads are controlled using an experiment control unit (ECU) PIC. The MM constantly measures the samples' properties and, therefore, implements a dedicated controller not to block or interrupt the CPD and DLP operations. The magnetometer was already described in the scope of the ADS subsystem because it serves a dual purpose of attitude determination and scientific data generation.

Another secondary payload consist of a GNSS receiver, which was included to fulfill two purposes:

1. To increase the spatial and temporal localization of other payload measurements.
2. To investigate the GNSS signal disruption, total or partial, due to ionospheric varying conditions.

The FireFly GNSS receiver with an active antenna model TW2406 (for GPS/GLONASS L1 band) were selected. This GNSS receiver was selected due to its low size (17.0 by 22.4 mm) and low power consumption (150 mW). It is considered one of the world's smallest space-capable GNSS receivers according to the manufacturer's information (http://www.sensorcomm.co.jp/catalog/firefly_flyer.pdf). This receiver has been used on-board small rockets launched by JAXA and other university satellites such as the Fireant Cubesat [21]. The receiver was included within the same PCB as the MM to optimize the space available in the top of the spacecraft, and to minimize the number of PIC microcontrollers. However, due to the lack of resources in the MM PIC microcontroller to handle both the MM experiment and the GNSS readings, and due to the low priority assigned to this secondary payload, the GNSS receiver has not been used. Therefore, it is not mentioned any further in this paper.

Table 1. Main payload instruments.

Item	Instrument/Sensor	Measurement Range
Charged particle detector based on CMOS sensors	6 detectors for protons, electrons and ions. 2 x-ray detectors	The final range will be defined based on ground calibration and orbital data.
Dosimeter for space radiation.	Liulin type dosimeter	Dose rate [Gy/h]: 1.12×10^{-8} to 0.188 Flux [particles/cm ² /s]: 0.01–10,000 Deposited Energy [MeV]: 0.081–20.833
Material mission	Strain measurements Temperature measurements	Strain: $10,000 \times 10^{-6}$ to 1×10^{-6} Temperature [C]: −55–150
DLP	Electron density/temperature	Current [A]: $\pm 10 \times 10^{-6}$ Minimum current [A]: 80×10^{-12} Voltage [V]: ± 11
Magnetic field measurements	Magnetometer based on 3-axis magnetoresistive sensor	$\pm 2 \times 10^{-4}$ [T]

2.4. Ten-Koh General Arrangement

Figure 6 shows the arrangement of the subsystem assemblies in the Ten-Koh structure. Most of the subsystems parts, including the payload instruments control electronics and several instruments, are mounted on the internal structure.

The material mission (MM) payload control electronics circuits are located inside the structure, while the material samples are placed directly above and exposed to the space environment. Similarly, the double Langmuir probe (DLP) electrodes are located on the outside of the external structure of the satellite. However, its control electronics circuits are housed in the internal structure shielded in a 3 mm-thick aluminum box.

The magnetometer instrument is mounted on the top panel in an aluminum box. The selection of the location and materials for the mechanical interface of the magnetometer instrument is discussed below as part of the challenges in the design of the Ten-Koh satellite.

2.5. Satellite Concept of Operations

For all experiments and mission objectives, the concept of operations is shown in Figure 7. There is one main ground station (MGS) located at Kyutech, from where all missions are planned and executed, and housekeeping and payload data are downlinked. The mission was defined to start operations in a period of 2–3 weeks after the launch by following the next sequence of events:

1. Launch on 29 October 2018 at 13:10 Japan Standard Time (04:10 UTC) from the Tanegashima Space Center in Japan, onboard the H-2A rocket number F40 as a piggyback payload of the GOSAT-2 satellite.
2. Separation from the rocket 2000 s after liftoff at an altitude of 623 km and with an orbital velocity of 7.5 km/s. The preliminary orbit was estimated to plan tracking of the satellite during the first hours after separation, until a more precise ephemeris was obtained later: semi-major axis of 6969.772 km; eccentricity of 0.002586; inclination of 97.835 deg; argument of perigee of 27.546 deg; longitude of the ascending node of 313.285 deg; and true anomaly of 212.990 deg. All data were provided by the Japan Aerospace Exploration Agency (JAXA) prior to the launch [22]. Nominal operations were carried out using Two Line Element sets (TLEs), as customary for small satellites.
3. After the separation and orbit insertion, and once all systems onboard were verified, the mission operations started on 19 November 2018.
4. During each communication session with the main ground station located in Kyutech, housekeeping telemetry and mission data are downlinked and missions are executed according to a previous plan (see the following process of mission planning and execution in Steps A–D).

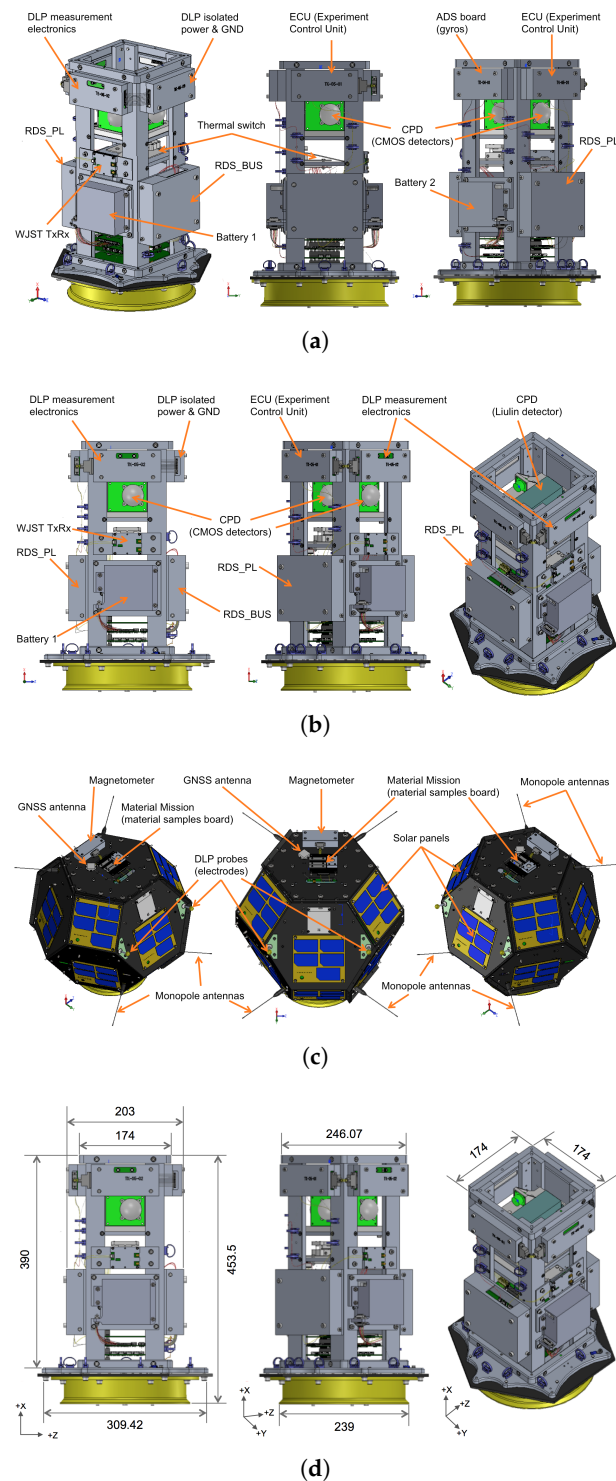


Figure 6. The payload arrangement onboard the spacecraft: external arrangement (a); internal arrangement (b,c); and internal structure dimensions in millimeters (d).

Based on the above key events, the mission operation phases given in Table 2 have been defined. Each phase refers to a section of the life cycle of Ten-Koh satellite, which itself starts with the launch and lasts until the decommissioning of the mission.

The process of performing all missions on-board Ten-Koh (Processes 3 and 4 in the concept of operations above) is as follows:

- A An experiment is planned to be executed in a selected position and epoch along the orbit with a specified time duration.
- B A set of uplink commands is configured and scheduled for commanding the satellite in a selected pass over the MGS in Japan.
- C During the designated pass for operations, the satellite health status is confirmed first by housekeeping telemetry. Then, the uplink command is received by the satellite and the OBC configures the payload instruments and the required subsystems to provide the resources for the mission to be executed.
- D The mission experiment is executed and the housekeeping and missions data are stored in different memories on-board the satellite to be downlinked during the next passes over the MGS.

The process in Steps A–D is repeated for every new mission experiment.

Table 2. Satellite mission operation phases.

Phase Number	Mission Phase	Duration	Description
1	LEOP: Launch early operations	2–3 weeks	- Starts after separation from the launch vehicle. - Subsystems and instruments are powered on. - Comprehensive satellite subsystem checkout. - Recording of Satellite Telemetry Data (STD). - Downloading and analysis of scientific and technical data by ground control station. - Checking the operation of payload instruments (by real time mode commands). - Transition to normal operation.
2	Normal Operation	1 year starting after 3 weeks from the launch	- Science data collection. - Long and short term planning of payload instruments. - Communication sessions with ground stations in charge of controlling the satellite.
3	Extended Operation	1 year, starting after Normal Operation	- Variations in science operations. - Orbit decay monitoring.
4	End of mission	>3 years	Planning and execution of the end of life operation.

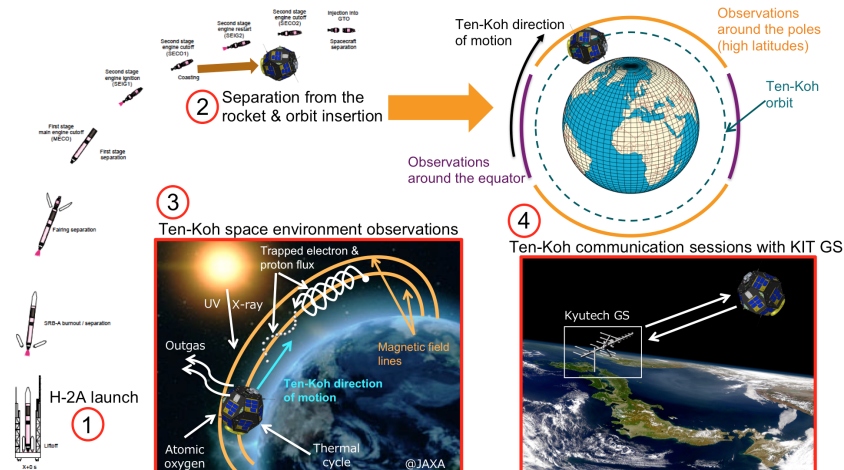


Figure 7. Satellite concept of operations as explained in sequence from 1 to 4. In Processes 3 and 4, a mission is configured to be executed and later payload and housekeeping data are downlinked. This mission configuration–execution–downlinking occurs following the execution of Steps A–D. After launch and when the nominal operations started, only Processes 3 and 4 are present as part of the nominal operation.

2.6. Design of Mission Operations

The Japanese Aerospace Agency (JAXA) offered free launch slots for the GOSAT-2 launch in early 2017, and the Ten-Koh mission design was based on general estimates and assumptions for a Sun-synchronous polar orbit with an altitude of 613 km and 97.8 degrees inclination (the final orbit parameters were not confirmed until a few months prior to the launch). The previously described experiments and the associated six instruments were conceptualized during a brainstorming process and an official proposal was submitted to the JAXA as a bid for the GOSAT-2 launch opportunity, which was offered to Japanese universities. After the launch, the final orbit was confirmed to have the following parameters:

- Perigee: 593 km.
- Apogee: 615 km.
- Inclination: 97.8 degree (retrograde orbit).
- Semimajor axis: 6975 km.
- Orbital period: 96.6 min.
- Eclipse duration: 47.7 min.
- Number of revisit times per day over Kyutech (Japan): 4.
- Revisit cycle: 9 days.
- Local equator crossing time of descending node: 13:00 h.

Even though Ten-Koh can pass over the Kyutech ground station in Japan up to four times per day (twice at around 13:00 and twice at 01:00), only passes with more than 15 degrees of elevation are used. This is done in order to establish reliable communication between the ground station and the satellite (i.e., to avoid the loss of data packets). Ultimately, an average of two passes per day are usable.

Satellite Operation Modes

The operation of the satellite is divided in the following operation modes. These are linked to the specific events from the concept of operations previously explained in Figure 7 via the number shown in parenthesis:

- LEOP mode (1).
- Communication sessions mode (4).

- Telemetry mode (4).
- Real-time mode (4).
- Normal mode (2).
- Payload operation mode (3, 4).
- Satellite log mode (4).
- Direct command mode (4).
- Set SD card saving data address mode (4).

During the payload mode, the MGS sends a command or a series of commands for executing payload operation according to the following mission experiments:

- CPD: measurements of CPD are sampled and stored it in the ECU's SD card.
- DLP: measurements of DLP are sampled and stored it in the ECU's SD card.
- MM: measurements of MM are sampled and stored it in the MM's SD card.
- ADS: measurements of ADS (magnetometer, gyroscopes and Sun sensors) are sampled and stored it in the ADS controller's SD card.
- UCP experiment: measurements of UCP are sampled and stored it in the UCP's SD card.
- Thermal switch: measurements of the thermal switch are sampled and stored it in the UCP's SD card.

Depending on the payload instrument to be operated, the configuration for the operation must be included in the uplink command for the experiment to run. For the cases of CPD, DLP and ADS, the payload mode uses a definition of two operating windows along the orbit, which allows for the CPD and DLP cases to sample data from each instrument in two different locations of the satellite orbit. For instance, the two missions can be performed over the north and south poles, or in two opposite regions over the equator, as shown in Figure 8.

2.7. Challenges Faced during the Project

The Ten-Koh satellite was developed to meet the mission objectives specified in Section 2.1 while meeting constraints such as the size and mass limits, the orbit driven by the launch's primary payloads, the schedule dictated by JAXA, and the available resources and small team size. Especially the schedule and resource constraints have driven some of the key decisions throughout the project, both technical and programmatic in nature. The main challenges and solutions to them are summarized in Table 3 and then described individually in more depth.

2.7.1. Available Generated Power

The minimum available power in the worst-case scenario was computed to be 11 W, which was found sufficient for the maximum expected power consumption of 9.5 W. The EPS and the batteries were sized to be able to support such power consumption in eclipse for up to 30 min, which was enough to gather CPD data around the poles, regions of specific interest due to abundance of charged particles. The EPS was not sized to be able to support maximum power consumption indefinitely due to the limited surface area that could be covered with solar cells. Using a deployable solar array could remedy this area constraint but would require development of complex on-board software and mechanisms, and was thus deemed unfeasible. Attitude control would also be needed if a deployable array was used, which is described in a separate section.

Table 3. Design challenges and adopted solutions.

Challenge and Its Origin		Solution
a	<i>Challenge:</i> Available generated power <i>Origin:</i> SmallSat constraint	Maximized solar array area + payload operations with reduced time duration.
b	<i>Challenge:</i> Available space location, mass and structure interface for some payload instruments <i>Origin:</i> SmallSat constraint	Platform subsystems were compactly accommodated at the bottom of the central internal structure and the inside walls of the external structure to maximize payload space. Structure effects on the measured data need to be corrected for during result interpretation.
c	<i>Challenge:</i> Amount of data generated by the mission instruments <i>Origin:</i> SmallSat constraint	On-board storage capacity of up to 2 GB for the main payload instruments, EPS and ADS subsystems each.
d	<i>Challenge:</i> Structure design heritage <i>Origin:</i> Schedule for delivery on time	Shinen-2's legacy structure with location for Ten-Koh components, considering thermal control, fields of view of antennas and sensors.
e	<i>Challenge:</i> Attitude determination and control vs attitude determination <i>Origin:</i> Schedule for delivery on time and keeping the CPD and magnetometer readings with reduced influence	An attitude determination only approach was adopted to meet schedule and cost constraints.
f	<i>Challenge:</i> COTS design heritage from previous missions <i>Origin:</i> Reliability and budget constraints	Use of avionics components from Shinen-2 and other small satellite missions wherever possible.
g	<i>Challenge:</i> Magnetic measurement system with no deployable boom <i>Origin:</i> Schedule constraint	Magnetometer instrument location was designated in the top panel of the satellite to reduce complexity from boom deployable system while keeping the schedule and measurements quality.
h	<i>Challenge:</i> Launch date fixed by JAXA <i>Origin:</i> Schedule constraint	Adopted a modular design approach, with re-use of the same microcontroller and other circuitry in all the subsystems. This also maximized software reuse.

2.7.2. Available Space Location, Mass and Structure Interface for Some Payload Instruments

The payload instruments were located in the upper half of the satellite in order to provide an unobstructed view for the CPD particle detectors, expose the material samples to the space environment, and allow CPD and DLP to be controlled by the same ECU, thus allowing synchronized measurements. Initially, controlling the GNSS receiver and the magnetometer from the same microcontroller was foreseen but was eventually found unfeasible due to PIC16F877 hardware limitations. However, these payloads were kept in their original locations to avoid large configuration changes late-on in the project. Only the microcontroller responsible for these two instruments was changed.

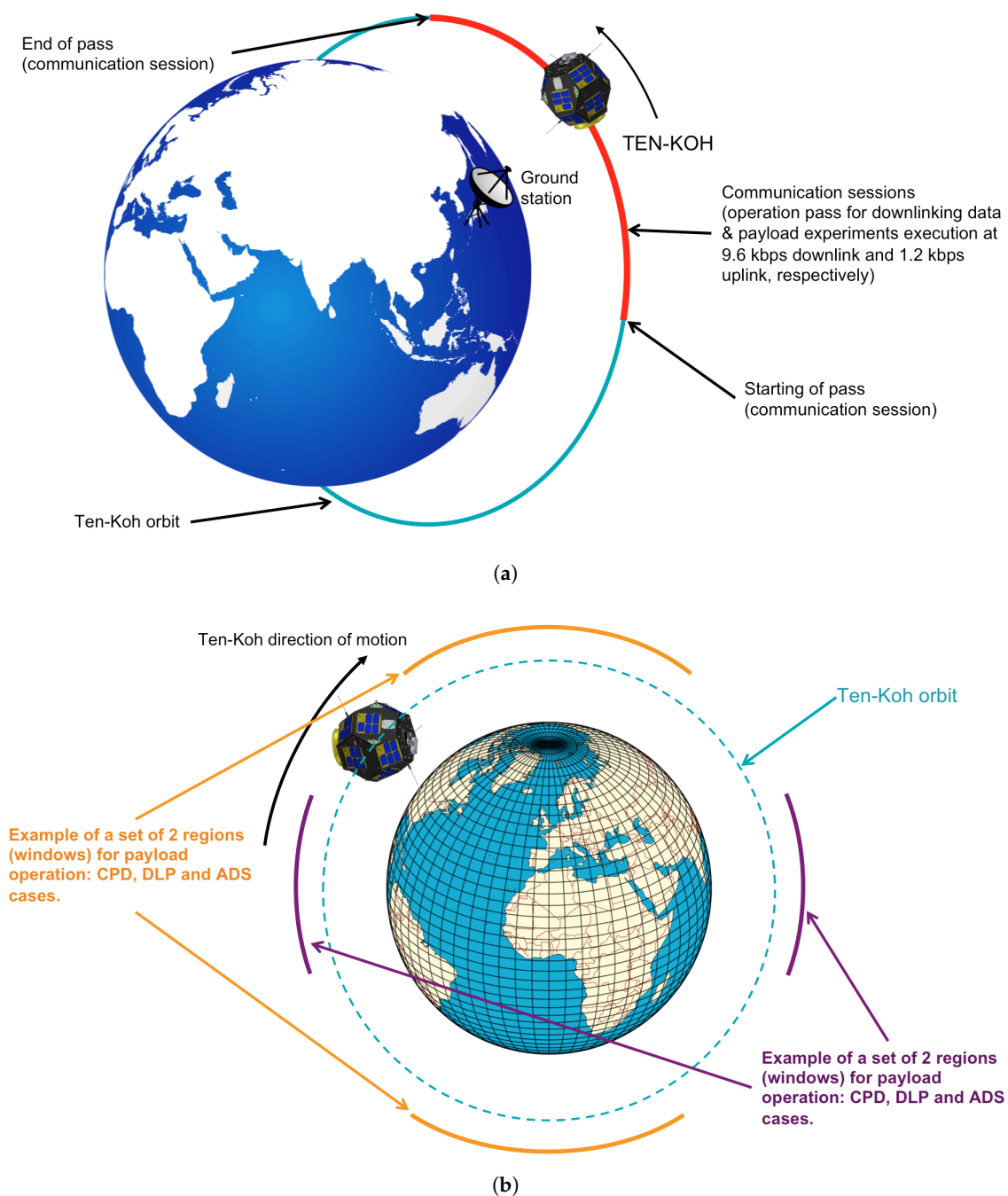


Figure 8. Operation of the main payloads as described in Processes 3 and 4 in Section 2.5. (a) Communication sessions between the satellite and Kyutech ground station in Japan occur in two periods, during the day at around 13:00 and during night time at around 01:00. (b) Mission execution for CPD, DLP and ADS in a specified region that is pre-selected and configured via a command during the communication sessions with the satellite.

In the case of the satellite's top panel, all the space available is dedicated to accommodate the material mission assembly, the magnetometer instrument and a GNSS receiver antenna. The magnetometer was placed on the top panel, as far as possible from other subsystems and particularly the switching voltage regulators in order to reduce the noise in the magnetic field measurements (explained further in section h).

The DLP electrodes were located on opposite sides of one of the hexagonal panels close to the ECU, with their exact location and length tailored so as to comply with the allowed physical envelope

of 50 cm × 50 cm × 50 cm dictated by the HII-A rocket. Due to this physical envelope limit and in order to avoid time-consuming development of an expandable electrode system, the DLP electrodes have fixed length. However, when fixing the electrodes close to the satellite structure, the measurement system must be able to measure plasma properties when the electrodes are located inside and outside of the plasma sheath, which implies the ability to measure exceedingly low current values. The IRI-2016 model [23,24] at the time of Ten-Koh launch (2018) was employed to find the expected plasma density in the range from 1.0409×10^{10} to $2.0248 \times 10^{11} \text{ m}^{-3}$ (mean $7.0153 \times 10^{10} \text{ m}^{-3}$), and an electron temperature of 2407.8 K (0.2 eV). The resulting Debye length of 1.28 cm suggested that the chosen probe length would indeed locate the probes inside and outside of the sheath, depending on spacecraft attitude (schematically shown in Figure 4). Thus, the size of the probes was found appropriate to address primary Primary Objective 1 and the measurement circuit was designed and tested accordingly, as shown in Section 3.1.

2.7.3. Amount of Data Generated by the Mission Instruments

Ten-Koh has been equipped with four SD cards of 2 GB each in order to provide on-board storage for the mission, ADS, and housekeeping data. The DLP instrument generates the largest amount of data per single measurement with 3000 bytes in the high-resolution mode, followed by the CPD with 2528 bytes (2000 from CMOS detectors and 528 from the Liulin detector). Table 4 shows the main mission instruments listed in descending order for the amount of data generated considering the different mission scenarios. Storing data on-board presents no difficulties due to the use of SD cards. However, the downlink of relatively large amounts of data can become problematic when using only the amateur UHF frequencies at the rate of 9.6 kbps supported by the used COTS transmitter. Thus, the experiment duration has to be tailored depending on team priorities and operational conditions. Increasing the number of downlink locations would improve the system data throughput at the expense of more energy needed per-orbit to operate the transmitter for longer. Ultimately, this approach was not adopted due to insufficient time to prepare additional ground stations before the launch. During the first three months of operations, maximum duration of 10 min was planned for each of the main missions. Depending on the real downlink capability, extending the duration for the main missions would be considered.

Table 4. Data generation for the main mission instruments as a function of the duration of each experiment for one ground station access case.

Mission Instrument	Amount of Data Generated Per Single Measurement Per Channel [Bytes/ch]	Maximum Time of the Experiment [minutes]	Data Generated for the Maximum Time Duration for 2 Channels [Bytes]
DLP	3000 bytes (hi-res)	15 min	900,000 bytes (hi-res)
	1500 bytes (med-res)		672,000 bytes (med-res)
	600 bytes (low-res)		180,000 bytes (low-res)
CPD	2528 bytes	15 min	75,840 bytes
ADS sensors	64 bytes	15 min	57,600 bytes
Material mission	60 bytes	5 min	18,000 bytes

2.7.4. Structure Design Heritage

Using a heritage Shinen-2 structure had the advantage of reusing the same composite mold and thus lowering the total development cost. In addition, it increased the confidence in the ability of the structure to withstand the launch environment. However, the arrangement of the subsystems in the structure has been changed completely in order to reduce the amount of harness as well as to provide the required space and viewing conditions for the payloads. The location of the DLP electrodes is described in Section 2.3, Figure 4, and Figure 6c. Besides that, the fields of view of the

CMOS sensors were kept as unobstructed as possible by placing the CPD in the center of the internal structure, sufficiently below the upper aluminum boxes of the ECU, DLP electronics and ADS.

Both Shinen-2 and Ten-Koh structures share the same shape and size but the material is different: Ten-Koh uses CFRP (carbon fiber + epoxy resin), while the external structure of Shinen-2 was based on CFRTP/PEEK material. CFRP/epoxy composite materials have been tested for their use in space, where the atomic oxygen and UV radiation represent the biggest challenge for graphite epoxy composites [25]. In this regard, CFRP/epoxy has been shown to have increased durability in LEO compared with CFRTP/PEEK. Both CFRTP and CFRP have similar radiation permeability [26], and thus would both provide similar shielding to the electronics and the CPD detectors. During the structural-thermal model (STM) tests, it was confirmed that the spacecraft can be qualified for the vibration environment of H-IIA (more than 100 Hz in the longitudinal direction, with -6.0 G compression and $+5.0$ G tensile, and more than 50 Hz in the lateral direction, with ± 6.0 G of compression and tensile).

2.7.5. Attitude Determination and Control vs. Attitude Determination

Performing only attitude determination without attitude control was chosen in order to reduce the satellite development time. Only passive magnetic attitude control was initially deemed feasible for schedule reasons and was proposed early in the design process. However, it would degrade the CPD and magnetometer readings and thus was discarded. From the point of view of the mission objectives, being able to understand the direction in which the particle/plasma measurements were taken was sufficient.

2.7.6. COTS Design Heritage From Previous Missions

Ten-Koh avionics are based on Shinen-2's COTS components and other Kyutech satellite missions that have been flight-proven in space. However, in certain cases, no heritage components from previous Kyutech missions were found. In these instances, literature review was conducted to find candidate COTS components. ADUM1250 and ADUM14xx families of magnetic isolation devices are of note in this respect because the entire satellite data handling subsystem relies on these ICs, and thus choosing components likely to survive space environment was needed [16]. The ADUM1250 IC is used for I2C isolation purposes while the ADUM14xx family is used for SPI and general-purpose digital signals with 5V-TTL compatible levels. If no components with reported flight heritage were found, device packaging, used technology, and temperature range were considered in order to select the IC from amongst the commercially-available ones.

2.7.7. Magnetic Measurement System With No Deployable Boom

The magnetometer instrument was installed on the top structural panel of the satellite next to the material samples (see Figure 6). Calibration was performed in order to validate whether the instrument was able to perform magnetic field measurements inside the expected range along the satellite orbit, and that it was not influenced beyond acceptable levels by the spacecraft self-induced magnetic noise. A sensitive and linear output inside the expected range was obtained, while confirming that the induced noise (offset from the spacecraft) was an order of magnitude below the instrument's range limits of $\pm 2 \times 10^{-4}$ T ($\pm 2 \times 10^5$ nT). Because the different offset values were constant (see Figure 9), it was possible to characterize the different values produced by the different modes of operation of the satellite, thus enabling the removal of magnetic noise from the measurements during post-processing of the data. This calibration and post-processing will be described in detail in a dedicated publication because it requires extensive explanation that cannot be accommodated in this paper.

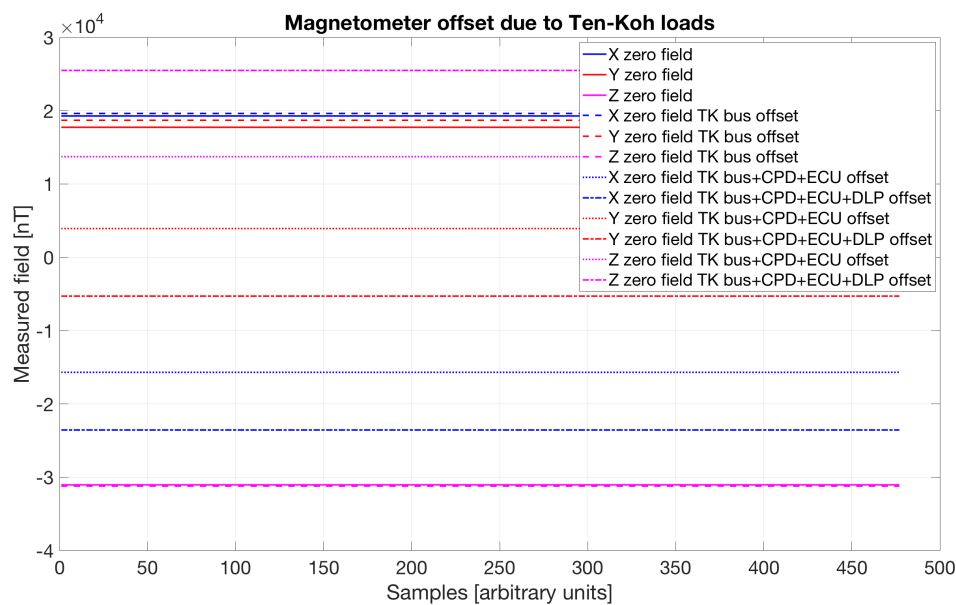


Figure 9. Magnetometer calibration tests. Offset imposed by the satellite to each axis when: no electronics are ON, and under specific subsystems operating conditions. The results will be used to remove the offset to the magnetometer measurements when the satellite is under specific operating conditions.

2.7.8. Launch Date Fixed by JAXA

The satellite launch slot was awarded to the team through a competition hosted by JAXA. However, to be eligible for the launch, the satellite had to be delivered and qualified for flight on-time with no possibility of postponing the launch, which would take place in case of primary launch payloads.

To meet the rigorous schedule, the system was kept as simple as possible: for example, no deployable boom for the magnetometer and no attitude control were used, as described above. Moreover, a modular design approach was adopted, where the same microcontroller and other circuitry were reused in all the subsystems. This also maximized the possibility to re-use the same lower-level software across the subsystems. Lastly, a proto-flight approach was adopted for system-level thermal-vacuum tests in order not to lock away the engineering qualification model in a vacuum chamber at the peak of the development activities.

3. Results

3.1. Ground Tests

The main payload instruments were tested and calibrated to verify that the experiments can provide measurements inside the defined range. For the MM, DLP and magnetometer, the testing and calibration were carried out using the facilities of Kyushu Institute of Technology in Japan. For the CPD case, which includes the CMOS and Liulin detectors, the tests were conducted at Prairie View A&M University and the Space Research and Technology Institute of the Bulgarian Academy of Sciences, respectively. Details about the Liulin detector calibration can be found in [27], while the CMOS detectors still need an extensive calibration that is currently underway at Prairie View A&M University.

Double Langmuir Probe (DLP)

Figure 10a shows the ground test results of the DLP plasma measurement system. The different I–V curves for each case correspond to varying numbers of samples, and hence the speed at which the applied potential was being changed. This resulted in different coupling of the circuit with the RF

plasma source in the chamber. For a large number of samples, the DLP generates smaller voltage steps (step size). Considering that the sheath formation varies as a function of the biasing voltage applied to the probes (V_B) with respect to the floating potential V_f (zero net current or equal flow of electrons and ions), by setting the probes potential above the V_f , a net positive current due to plasma electrons flows into the plasma. If V_B is set below V_f , then a net negative current of ions flows out from the plasma. By biasing the probes with negative and positive potentials, the typical I–V curve is obtained (see left of Figure 10a). After different tests performed at the LEO plasma chamber located in the facilities of the Laboratory of Spacecraft Environment INTERaction Engineering (LaSEINE) in Kyutech, the -10 to $+10$ V biasing range was deducted as optimal to obtain a full I–V curve in orbit. This range allows computing the electron density and temperature of the plasma surrounding the spacecraft according to the tests and estimated parameters from the IRI 2016 model [24] for the referred epoch.

When biasing the voltage across the probes in smaller steps (more samples), the DLP system takes longer time to sweep the voltage from -10 V to 10 V. When selecting a bigger step size (fewer samples), the DLP sweeps the voltage across the probes faster. This can be seen as a change in the frequency of the biasing voltage. Because the plasma source employed in the testing was of RF origin, the plasma potential fluctuated at the RF frequency and its harmonics. The overall effect appears as a wider I–V curve that creates a higher value of the electron temperature and shifts the floating potential to a more negative value, as described in [28]. To reduce the effect of the RF coupling with the measuring circuit, the selected step size used during the testing was between 500 and 1000, which corresponds to 65 and 131 samples. This was validated with the use of a source-meter instrument with an RF compensation stage and requesting the same number of samples.

The final ground test for the DLP instrument development was the calibration of the system. The fastest way to perform calibration of the measuring unit was by the use of a relative calibration with the same SMU2400 instrument. By comparing resistance measurements from current and voltage readings, it was possible to obtain the sensitivity and linearity of the DLP system relative to the SMU2400 and the resistor value. For this purpose, a resistor of $1\text{ G}\Omega$ (1×10^9 ohms), and 1% precision value, was connected between the two probes at room temperature. Then, the voltage was biased from -10 V to 10 V across the resistor. During this test, no plasma environment conditions were applied. Figure 10b,c shows the results from the calibration of the DLP instrument while Table 5 includes the calibration results for the four channels in the DLP system.

Results from the ground tests in the LEO plasma chamber show that the Ten-Koh DLP is provided with an electron temperature of 1.38 eV and a plasma density of $1.91 \times 10^{12}\text{ m}^{-3}$. As a reference, the LEO plasma chamber produces plasma densities in the range of $10^{11} - 10^{12}\text{ m}^{-3}$, meaning that the DLP results were plausible.

Table 5. Calibration results for all four channels of the Ten-Koh DLP instrument.

Channel	Measured Resistor Value [$\text{G}\Omega$]	Relative Error from 1 [$\text{G}\Omega$] Resistor Nominal Value in %	Relative Error from SMU Measured Resistance in %
1 (CHA1)	1.0010×10^9	0.1	0.82
2 (CHB1)	1.0180×10^9	1.8	2.53
3 (CHA2)	1.048×10^9	4.8	5.5
4 (CHB2)	9.944×10^8	0.56	0.15
SMU2400	9.929×10^8	0.71	—

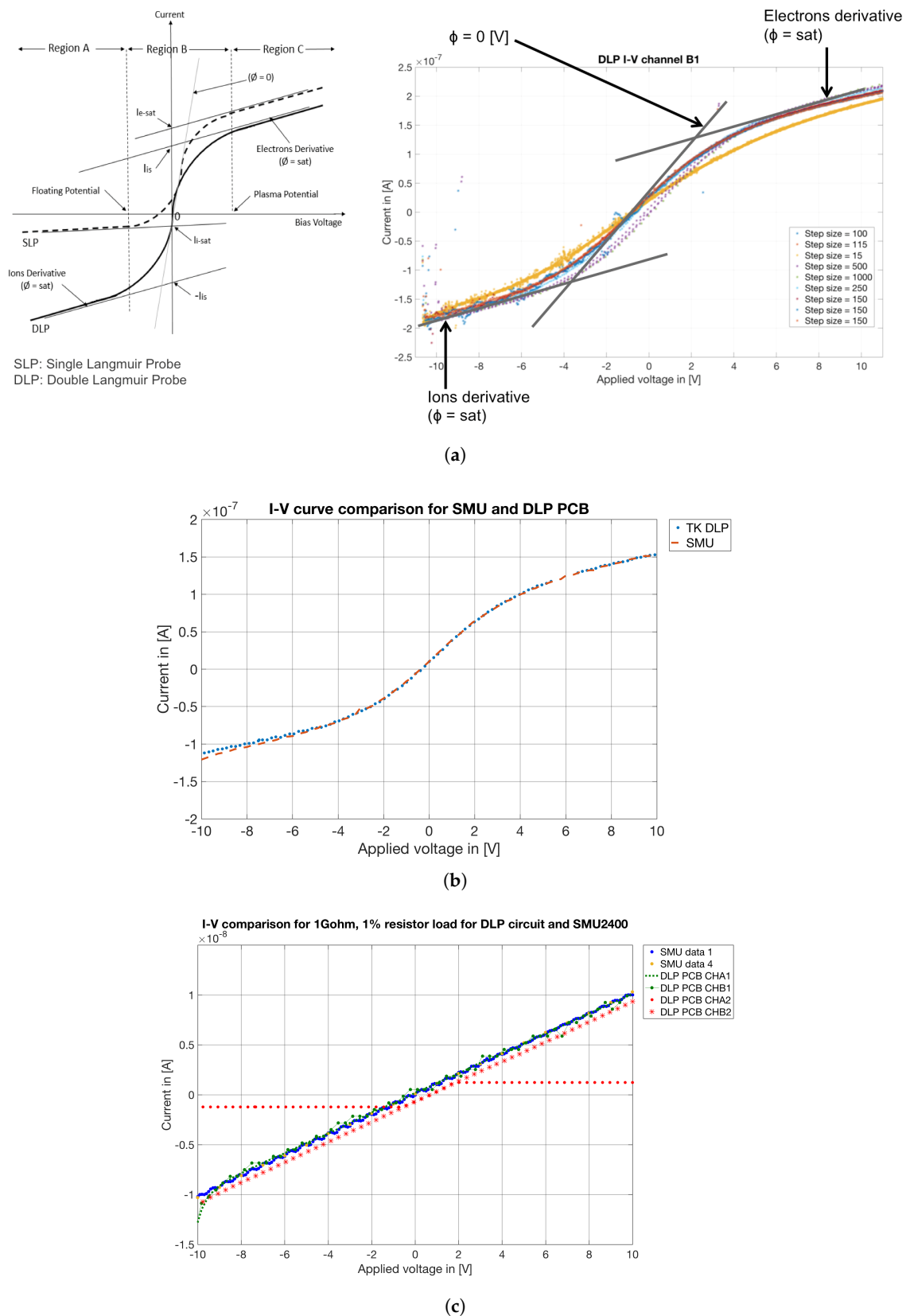


Figure 10. Ground tests results for: (a) DLP I-V curve results for different number of samples (DAC step size) and the results comparison to an ideal DLP I-V curve on the left; (b) DLP measuring circuit comparison with the SMU for the same plasma conditions inside the LEO plasma chamber; and (c) DLP sensitivity and linearity relative calibration by using a 1 G Ω resistor and the SMU2400 instrument. The DLP system has a linear response as expected inside its operating range. The current saturation in channel CHA2 is due to its higher gain compared with the other channels.

During the ground test of the DLP system, the EM satellite structure and the DLP electrodes mounted on it, as well as the plasma source were kept static inside the plasma chamber. The face of the satellite structure where the electrodes were mounted was positioned toward the direction of the plasma source (0 deg) inside the chamber and data were acquired. Then, after removal of the plasma and vacuum, the satellite structure was rotated by 180 degrees, i.e., the electrodes were facing away from the plasma source. Another set of measurements was acquired in this configuration. Only these two tests were conducted in order not to occupy the plasma chamber for longer than absolutely necessary, due to the high demand for the facility. However, the two tested orientations, with the DLP electrodes facing towards and away from the plasma flow, captured the two extreme thicknesses of the sheath in which the experiment would operate. This was associated with a reduction in the magnitude of the measured current in an order of 10^{-1} A.

Under flight conditions, the lack of attitude control invariably affects the results as the alignment of the DLP electrodes with respect to plasma flow varies continuously. In this regard, attitude knowledge is critical to provide the location of each DLP sweep and to understand how quickly the attitude is changing. The sampling rate also plays an important role because sweeps with higher resolutions require more time for the data acquisition and, therefore, the attitude will change more during the sweep. Because the attitude and attitude rates were uncertain, the experiment was designed to be configurable by software via an up-link command. The parameters that can be set by software are:

- Resolution: The number of samples for each I–V sweep can be a set of up to 16 bits voltage increments. Three exemplar cases are presented in Table 4—different resolutions can be implemented depending on the satellite rotation conditions.
- Biasing voltage range: The biasing voltage range can be selected from -10 to $+10$ V (default), or be any other combination in that range. The only constraint is that the biasing voltage always goes from the most negative value to the most positive one.

Any combination of the mentioned parameters can be set for the DLP experiment to run. With this flexibility, it is possible to adjust the configuration of the experiment to avoid limiting the sampling rate and the charging conditions around the electrodes. However, as described in Sections 4.2.2 and 4.2.3, due to several failures, the DLP experiment could not be operated in-orbit.

3.2. In-Orbit Results

3.2.1. CPD Instrument

Five datasets were obtained from the Liulin detector during the early operations phase of the Ten-Koh mission. Table 6 shows the processed data from three different days when Ten-Koh satellite was flying over Japan. The Liulin detector was operated in a real-time mode, which means that the measured data were transmitted to the ground station without storing them on the SD card. These datasets represent the first in-orbit measurements from the CPD instrument. The real-time mode of the CPD was selected in order to test the system behavior when turning on the major load on-board.

The data obtained from the early operations show that, for close L-shell [29] values (1.129 to 1.206) and around the same time conditions (afternoon local time), the fluence has an average value of 0.28 particles/cm²/s. For a GEO field with no perturbations and short duration measurements period (less than 1 min), the particles flux is expected to be in this order, i.e., the CPD data are consistent with the expectations. However, the dose rate from the CPD data shows fluctuations. The average dose rate measured was 1.74 µGy/h, which is equivalent to 4.832×10^{-8} rad/s. The low dose rate measured in the second spectrum on 18 November 2018, comes from only low energy channels registering hits in the instrument. Figure 11 shows the five spectra and the corresponding average spectrum (in black) from the real time datasets. The horizontal axis in Figure 11 represents the deposited energy (in 256 channels) for the energy range from 0.081 to 20.833 MeV of the Liulin detector. In yellow, the spectrum with the lowest dose rate shows that the instrument registered only deposited energy in the low energy

channels, which in turn reduced the received dose. Similarly, the second spectrum on 18 November 2018, measured deposited energy in the low region of the instrument's energy range. In both cases, the total dose rate resulted in less than $1 \mu\text{Gy/h}$. The cause of the fluctuations in the dose rate is associated with the rotation of the satellite, which misaligned the CPD-Liulin detector with respect to the GEO field lines as it was traveling around the Earth. This is observed for the spectrum 2 on 12 November and 18 November 2018, where the precedent spectrum in each case measured a higher dose rate than spectrum number 2. Figure 12 shows the geographic locations of the real-time mode CPD mission performed on Nov 2018.

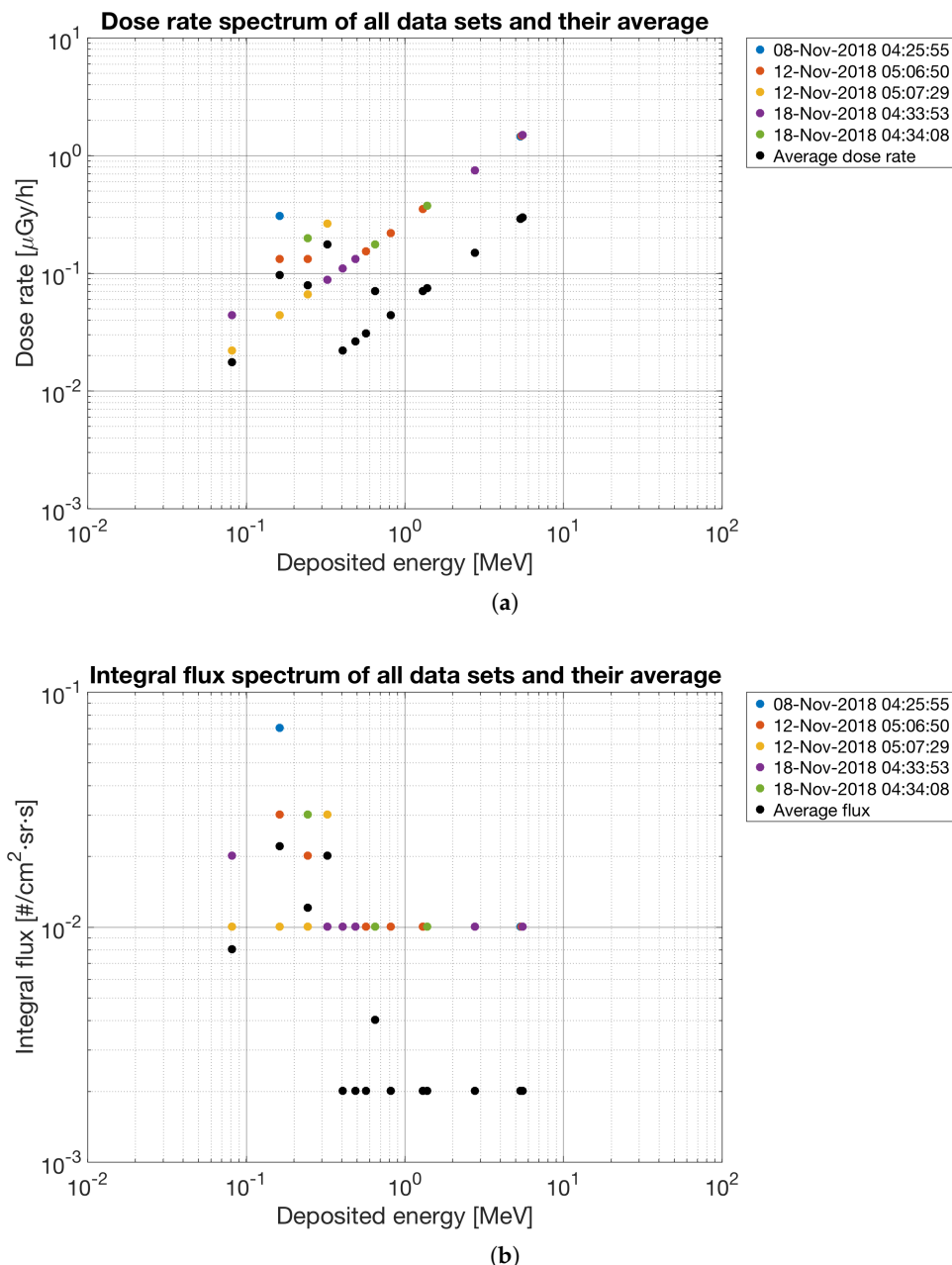


Figure 11. Dose rate and flux spectra obtained from the real time operation mode for the first CPD-Liulin data measurements on November 2018 during the early operation phase of Ten-Koh satellite: (a) dose rate spectrum for each dataset and the average spectrum measured inside Ten-Koh spacecraft; and (b) integral flux spectrum for the 5 real time datasets and the average flux spectrum measured inside the spacecraft.

Table 6. The Liulin spectra datasets obtained during real-time operations of the instrument. The measured dose rate and flux were obtained from a total exposition time of 15.286 s for each spectrum dataset. Epoch is in UTC even though the operations were conducted using the Japanese Standard Time (JST).

Spectrum	Epoch UTC	Geographic Location of the Satellite	L-Shell (McIlwain)	Dose [$\mu\text{Gy/h}$]	Rate	Dose Rate [rad/s]	Flux [$\text{particles/cm}^2/\text{s}$]
1	08 November 2018 04:25:55	Lat [deg]: 25.98 Lon [deg]: 132.67 Altitude [km]: 609.33	1.144	1.93		5.444×10^{-8}	0.33
1	12 November 2018 05:06:50	Lat [deg]: 29.55 Lon [deg]: 123.09 Altitude [km]: 608.41	1.206	2.66		7.389×10^{-8}	0.39
2	12 November 2018 05:07:29	Lat [deg]: 27.16 Lon [deg]: 122.50 Altitude [km]: 607.76	1.161	0.39		1.083×10^{-8}	0.2
1	18 November 2018 04:33:53	Lat [deg]: 26.04 Lon [deg]: 130.75 Altitude [km]: 604.32	1.143	2.79		7.750×10^{-8}	0.26
2	18 November 2018 04:34:08	Lat [deg]: 25.11 Lon [deg]: 130.53 Altitude [km]: 604.05	1.129	0.92		2.556×10^{-8}	0.23

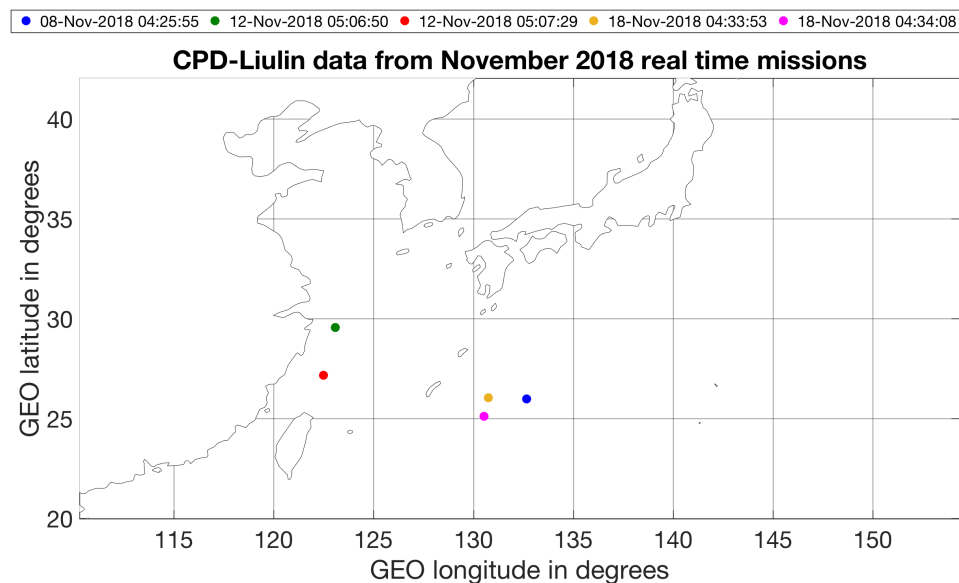


Figure 12. Geographic locations of the CPD missions conducted on November 2018.

3.2.2. DLP Instrument

During phase 1 of the mission (LEOP or early operations), the DLP was operated in the real-time mode similarly to the CPD. The data were transmitted to the ground station at Kyutech when the satellite was in view and after it had received the command for the DLP real-time operation. During these tests, the three DLP resolution modes were selected. However, in all cases, most of the received data packets contained zeros. Certain data packets containing data from the DLP instrument were received among the zero values packets. However, the received samples were not sufficient to reconstruct a single I-V curve. A problem with the DLP power line, which would prevent the instrument for switching on for the required duration was suspected. This is presented in more detail in the discussion section.

3.2.3. Magnetometer

During early operations, the magnetometer was also operated in real-time mode. The obtained data shown in Figure 13 demonstrate that even though the instrument has been mounted on the body of the spacecraft, the readings can be obtained with no significant noise coming from the other spacecraft subsystems. In Figure 13, the sinusoidal oscillations are a consequence of the spinning of the satellite. For attitude determination purposes, the magnetometer information is processed together with the Sun sensor and gyroscope data. As for the GEO field readings, the local field at the spacecraft position is obtained by using the calibrated model of the instrument. This calibration process and the comparison of the results with geomagnetic field models like the World Magnetic Model [30] have to be discussed in a separate publication in order to be able to explain it in sufficient detail to be reproducible. Further readings of the magnetometer were not always possible (the instrument returned only zeros), which is discussed below.

3.2.4. Material Mission (MM)

Data from the material mission have shown that it is possible to obtain information regarding the CTE change of each of the samples from strain and temperature sensors. A comparison with the preliminary ground tests and data measured in orbit for Sample 3 (described in Section 2.3) is shown in Figure 14. The data show that CTE does not remain constant and is dependent on the temperature changes as well as the carbon fiber direction.

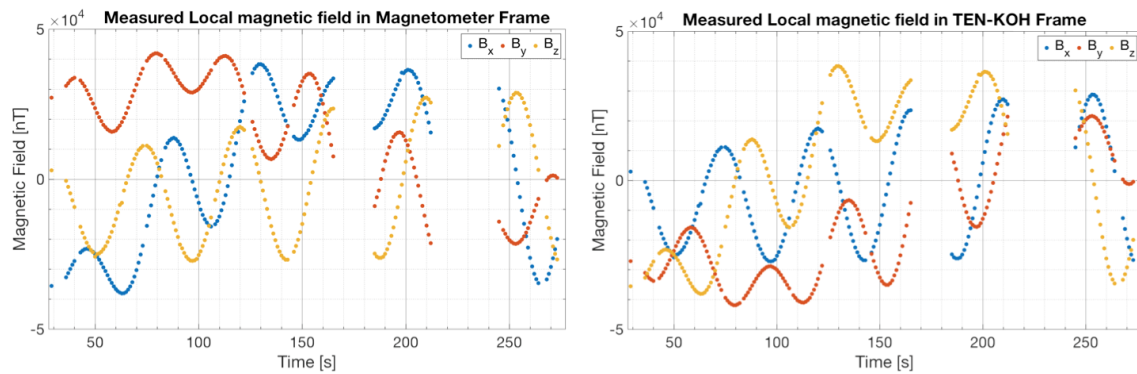


Figure 13. Magnetometer measurements obtained in the instrument frame of reference (**left**) and in the Ten-Koh frame of reference (**right**) for a period of 220 s. The discontinuities come from loss of data packets in the downlinking process during a communication session between the satellite and Kyutech ground station.

Strain was measured in two orthogonal, in-plane directions in each sample. While manufacturing the samples with embedded strain gauges, the alignment of the gauges with respect to the direction of the carbon fibers was also shown to modify the readings of the applied strain. The preliminary results from the MM appear to match the previous tests [31]. To have a complete understanding of the material degradation, a correlation of the orbital data with the ground calibration information will be performed. This will be included in a future publication regarding the material mission results.

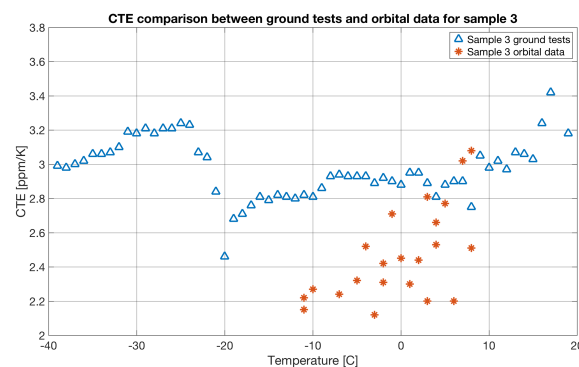


Figure 14. Material mission data obtained between December 2018 and January 2019. Estimated CTE from orbital data compared versus the ground test data for Sample 3.

4. Discussion

4.1. On-Orbit Operations History

After the launch, data started to be acquired according to the planned mission phases. The first action was the confirmation of the beacon signal reception in several parts of the world. After that, the operations from the Kyutech MGS started by testing all of the subsystems individually (LEOP phase) during the three weeks following the launch. In this period, the main payloads were turned on for a short time to verify the power consumption and that the different experimental data can be read from the different sensors in real-time mode.

During the LEOP phase, operation training and planning were conducted with the Ten-Koh operators team. For three weeks, the students involved in the operations developed procedures and scheduling tasks that were tested and verified first with the EM model of the satellite before being executed in orbit. This procedure of testing with the EM model allowed the correction of some ground station errors, before commanding the satellite. For example, an error in the pointing of the antenna was corrected to enable more accurate satellite tracking and improve the downlink data throughput.

Another corrected error refers to software bugs in the commanding interface application, which were solved by an upgrade of the software interface.

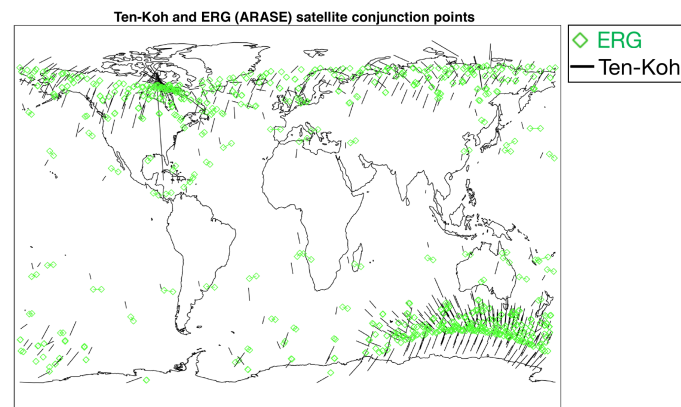
Once all ground systems were tested and ready, and after the confirmation of the operation status of the different Ten-Koh subsystems, the nominal operations phase started on 19 November 2018. The first main mission was to operate the CPD instrument for a total duration of 4 min 43 s.

CPD and ARASE Coordinated Observations

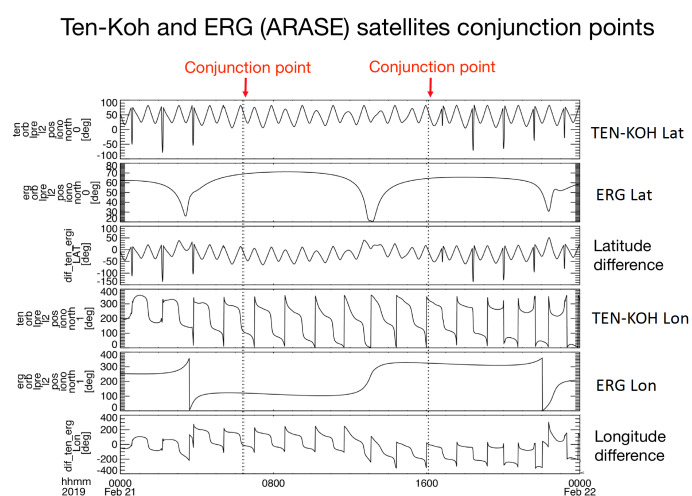
Because the CPD instrument is the largest electrical load on-board, using power with 8.2 W, its operating time was limited to at most 5 min twice every orbit. This high power consumption is exacerbated by the fact that the region of operation of the CPD is mainly located at high latitudes and around the polar caps, where the satellite can enter the eclipse region while operating the CPD. For the specific case of the first CPD mission, on 19 November 2018, the mission was executed over the north pole starting from a latitude of 82.2 degrees and a longitude of -165.28 degrees to a final latitude of 70.98 degrees and a longitude of 126.63 degrees, a zone that was inside Earth's shadow at that time. During this first mission, data from the power consumption and battery usage (discharge) were obtained, which allowed for the planning of longer duration CPD missions later.

In total, CPD observations were performed between 19 November 2018 and 19 March 2019, generating approximately 5.9 MB of data. The datasets from the first 12 missions (November to January) have been retrieved completely from the ECU SD memory, which is equivalent to 0.68 MB or 11.5% of the 5.9 MB. Because the battery and the power line that feeds the CPD have been performing well, the duration of the CPD missions was extended, with the longest one lasting a total of 100.5 min. This was equivalent to 201 individual readings from the CPD-CMOS and Liulin instruments. Table 3 shows that one of the challenges in the satellite development is the volume of data generated by the main payload instruments. The storage of such amount of data was addressed by selecting a storage device with 2 GB capacity. However, the communication bandwidth limitation remains the bottleneck when retrieving experiment data.

Based on the characteristics of the Ten-Koh orbit and the CPD detectors energy range, a collaboration with the ARASE (ERG) satellite [32] started from December 2018. Conjunction operations for the observation of high-energy electrons and ions were deduced feasible in locations when the difference between latitude and longitude of both satellites footprint at 100 km altitude is within five degrees. An example of the conjunction points between both satellites is presented in Figure 15. The ARASE team provided the conjunction points of both satellites by the use of the ARASE orbit prediction tool and the latest TLE available for Ten-Koh. Then, the Ten-Koh team produced the corresponding set of commands and schedule the time of operations for commanding the satellite. The satellite performed the joint observations with ARASE satellite when the on-board time matched the epoch specified in the uplinked command. The results from the cooperation with the ARASE satellite will be presented in a subsequent publication because they do not fall in the scope of this paper.



(a)



(b)

Figure 15. Ten-Koh and ERG (ARASE) conjunction points: (a) world map of the conjunction points for the period between 4 to 30 December 2019, showing both satellites footprints; and (b) example of two conjunction points computed for the period between 21 February 2019 and 22 February 2019.

4.2. Mission Operations Issues and Status

4.2.1. CPD Operation

The CPD was tested successfully during the LEOP phase of the satellite operations. The five datasets obtained from the Liulin detector over Japan were measured with an average exposition time of 15.2 s. In comparison, the exposition time for the normal mode missions was 30 s in average. The reduced exposure time caused the early operation (real-time) data to receive fewer counts, and less total dose rate and flux than the normal mode missions data. Variation in the dose and flux for the real-time data comes from two data series (12 November at 05:07:29 UTC and 18 November at 04:34:08 UTC) with most of the hits in the low deposited-energy channels of the detector. The main reason for the decrease in dose rate and flux in the mentioned datasets relates to the rotation of the satellite and the misalignment of the magnetic field lines with respect to the normal direction of the Liulin detector. The effect of the satellite rotation was perceived from the beginning of the operations during the communication sessions with the Kyutech MGS. The rotation of the satellite makes the beacon signal vary in intensity during communication sessions, which was observed during all communication session operations. This highlights the fact that schedule and resource constraints of the project have had a measurable effect on the scientific return of the mission. A similar pattern is

expected for other satellites that are launched as secondary (piggyback) payloads, without the ability to choose the launch date.

The Liulin results show an increase in dose rate and flux in regions where the ambient models predict the presence of electrons with more than 2.2 MeV and protons with more than 30 MeV. These energy values correspond to the energy required to penetrate the shielding of Ten-Koh external structure for each particle type. Had the instrument been placed with an unobstructed view of deep-space, lower energies could have also been registered. This, however, would have necessitated a complete redesign of the satellite structure that was not feasible from schedule point of view.

4.2.2. DLP Failure

A problem with the power line that feeds the DLP measurement system was encountered, making it impossible for the 5 V line to sustain its nominal output level. The root cause of the problem was identified as activation of the overcurrent protection due to the high-frequency transient from the inrush current of the DLP DC/DC converter. During the thermal-vacuum (TVAC) tests, the problem appeared for the first time in the EM model of the satellite. The issue was solved by reducing the amplitude of the inrush current through the use of an in-series resistor with the power line output from the EPS. However, once in orbit, it is suspected that changes in resistance of the used resistor (less than 1 Ω resistor value) triggered by temperature variations with respect to ambient conditions on-ground made the problem reappear.

Comparison tests performed with the EM and FM spare models of the DLP and the EM regulation board from the EPS showed a discrepancy in the inrush current of the DLP DC/DC converter between the EM and FM versions. Even though the difference in the inrush current from EM to FM translates to a difference of the in-series resistor of 0.3 Ω , it is high enough to trigger the OCP in the FM EPS DLP power line. During the calibration tests and the development of the flight software, the DLP instrument was powered by a workbench power supply adjusted to the same current limit as the DLP line from the EPS (1.5 A). The nominal current consumption was 0.27 A for both the EM and FM hardware. This caused the inrush current issue not to be identified at an earlier stage.

If re-work of the satellite were possible, a power regulation board (REG_PL) that accounts for the non-linear nature of different loads should be designed to avoid similar issues. As for addressing the issue of the unit currently in orbit, temperature variations will cause the in-series resistor to change its value. Therefore, it is possible that the DLP instrument could be operated at certain times during the lifetime of the spacecraft, as the orbit will naturally evolve and the thermal environment inside the satellite will vary accordingly.

4.2.3. SD Card and Magnetometer Readings Failure

The failures in reading the SD card and the magnetometer were caused by the controller board, which includes a single PIC microcontroller. The same PIC is used to read the telemetries of the 12 solar panels (voltage, current and temperature) that are also used for attitude determination purposes. This solution has been followed in order to synchronize the readings of all the ADS telemetries. However, managing so many devices with a single PIC strained the available hardware resources of the microcontroller. The PIC did not have in-built hardware support for all the data buses that it was required to operate simultaneously. Thus, I2C communication with the OBC was implemented making use of the in-built hardware in the microcontroller, while SPI communication with the sensors and the SD card was implemented in software.

Solar panel and magnetometer board ADCs are sequentially polled to keep their readings synchronized. The acquired data are either stored on the SD card or directly transferred to the OBC for near real-time downlink. When implementing the SPI communication protocol entirely in software, the timing of the clock and the data signals needs to be maintained in order to keep the system operational. Changes in the timing of these signals turned out to be critical when operating the sensor polling sequence, and have ultimately caused the entire SPI bus to suffer partial or complete

outages for prolonged periods of time. The changes in this timing could have been caused by either radiation effects, or by changes in temperature of the PIC and its local oscillator. Even though all of the ADS devices have been connected to a single SPI data bus, a propagation of a hardware failure of one of these devices can be ruled out because the devices have been isolated from each other.

The SD card failed after the first test during the LEOP phase with no possibility of recovering it later. The magnetometer could be read only once during the LEOP phase, and again after 4 March 2019. Between 4 and 17 March 2019, the magnetometer readings operated nominally in the real-time mode that did not make use of the SD card. However, operation in real-time mode means that the magnetometer could only be operated when the satellite was passing over the Kyutech ground station.

Only a power cycle of the system could restore the ADS operation, which was confirmed during ground tests. However, because the EPS and the ADS controller boards have not been designed to be powered off once the satellite began operating, this solution cannot be implemented in-orbit. All other subsystems can be power cycled, including all the payloads, both COMMS boards, and the OBC board. Even though the EEPROM in the OBC PIC has encountered errors during operation, the system has resumed nominal operations after a power cycle. In the case of the payloads, the ECU and MM controllers are kept off most of the time, and only receive power when an experiment is scheduled for execution. No errors have been encountered in these microcontrollers since the launch. Based on this microcontroller operation evidence, the lack of power cycle capability in the EPS and ADS boards made the microcontrollers of these subsystems more susceptible to failure, and limited the number of means to recover them.

4.2.4. OBC EEPROM Errors

Errors in the microcontroller EEPROM in the OBC were detected during the operation. The errors made changes in the log register of the commands processed by the OBC and their origin is suspected to be related to single event upsets (SEU). The OBC operation was always recovered by applying a power cycle from the EPS and then updating the on-board time of the satellite by a ground command. During the period of the last days of December 2018 and the first days of January 2019, more resets in the OBC were carried out than in the preceding weeks. However, none of these resets occurred during the execution of one of the main missions. Most of the errors and OBC resets in this period caused at most a change in the satellite time and the loss of the last received commands.

Besides the OBC EEPROM errors, similar errors were observed in other microcontrollers' EEPROMs. In particular, the EPS microcontroller experienced a change in the SD card memory address value stored in the EEPROM. This value is used by the EPS microcontroller to know the last used location of the SD card, where the most-recent housekeeping data have been stored. The solution for the EPS EEPROM issue was a regular update of the SD address for saving of the data following a ground command.

Commercial CMOS-EEPROMs have been shown to be susceptible to SEU while they are powered on [33]. Even though the HEF-EEPROMs (high-endurance flash) on-board Ten-Koh use a different technology, it is suspected that they might be sensitive to SEU in the same way. Because the microcontroller EEPROM is used to store commands and/or update parameters during the execution of the main software routine, changes in bits in the EEPROM could potentially alter and even stop the correct operation of the microcontroller completely. A detailed investigation of the radiation effects on the HEF-EEPROM of the PIC16F877 is being conducted and the results will be published in a subsequent publication.

4.2.5. Spacecraft Status

The Ten-Koh spacecraft was continuously operated daily since its launch (29 October 2018) until 19 March 2019, when a loss of communication between the spacecraft and the Kyutech ground station occurred. The ground-track of the last orbits of the satellite where the failure occurred are shown in Figure 16. The last data from its beacon were received over Greece on 18 March 2019 between 23:12:33

and 23:25:43 UTC. The decoded data indicated that all the main housekeeping parameters were inside their operating ranges, as shown in Figure 17. According to the same beacon data, the satellite was in the expected mission mode, i.e., nominal CPD mission scheduled between 22:00 on 18 March and 00:20 on 19 March UTC. The mission was supposed to finish when the satellite was coming from above the South Pole and heading northwest towards the southeast of South Africa. After finishing the CPD mission, Ten-Koh traveled for three more orbits, including two of them over the SA anomaly, before the pass over Kyutech ground station on 19 March 2019 at 13:31 JST (19 March 2019 at 04:31 UTC) when no signal was received. Irrespective of the anomaly that made the satellite fail, it occurred during the two orbits following the CPD mission and before the pass over the Kyutech ground station.

A review of the geomagnetic activity parameters from the NOAA Space Weather Prediction Center between 15 March 2019 and 18 March 2019 revealed that the Kp-index, an average of K-indices from a network of geomagnetic observatories, showed above the threshold value of 4 on 17 March 2019. As shown in Figure 18, Kp = 4 was observed between 21:00 on 16 March 2019 and 00:00 on 17 March 2019 UTC, and increased to Kp = 5 between 00:00 and 03:00 on 17 March 2019 UTC. Kp then decreased to Kp = 4 between 03:00 and 09:00 on 17 March 2019 UTC. The Kp-index with values bigger than 4 indicates a geomagnetic storm activity.

The Ten-Koh anomaly occurred approximately 50 h after the geomagnetic activity (between 18:00 on 18 March 2019 UTC and 03:00 on 19 March 2019 UTC). This shows a direct correlation between the anomaly and the presence of a geomagnetic storm. Space weather can produce temporal variations in the trapped radiation [34], which in turn can cause SEE and charging (surface and internal) that result in anomalies experienced by satellites.

The main hypothesis for the failure of the satellite was an error triggered by a single event effect (SEE), which was previously observed as errors in the in-built EEPROM of several microcontrollers. Note that such events were expected to take place over South Atlantic Anomaly due to presence of high-energy protons (in excess of 30 MeV) that could penetrate the spacecraft structure.

As in the case of the readings from the magnetometer that were stopped and then recovered after some time, the satellite began to transmit its beacon again after a solar storm that occurred on 10 May 2019 and that arrived at Earth between 14 and 16 May 2019. This correlation between and increased number of particles in LEO caused by the solar storm, and recovery of the satellite functionality corroborates the hypothesis that it was originally disabled by radiation damage. It is thought that the bit changes in the in-built EEPROM or microcontroller were undone by subsequent single event upsets so that the satellite could resume operations. It is also possible that the satellite was originally disabled by a single event latch-up, which did not increase the OBC power consumption enough to trigger the over-current protection. Subsequent latch-ups could have eventually caused the OCP trip, and the OBC to reboot and resume operations. This, however, is not the most likely scenario because multiple reboots were attempted from the ground but to no avail.

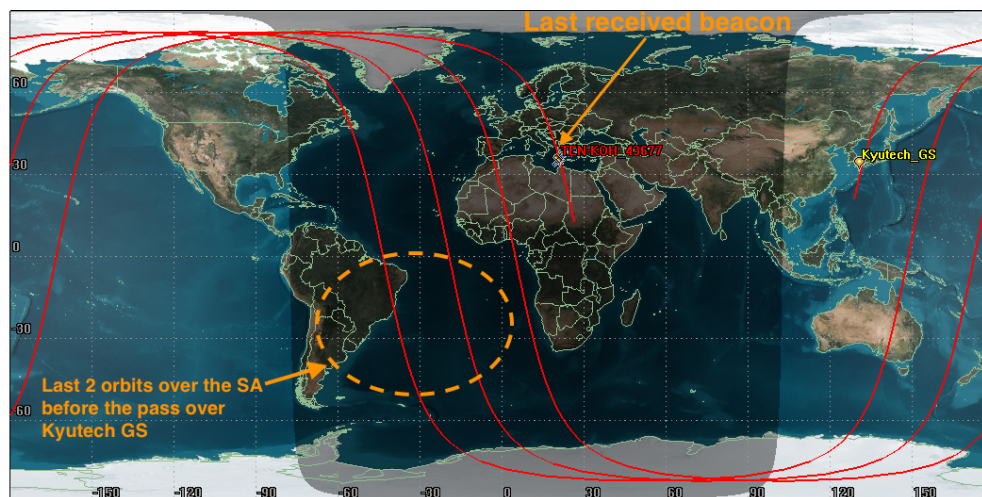


Figure 16. Ten-Koh orbit ground-tracks when the satellite is suspected to have failed. Following a beacon reception in Greece, Ten-Koh flew over the SA region in two consecutive orbits (shown in orange) before passing over the Kyutech ground station (yellow mark) in Japan on 19 March 2018.

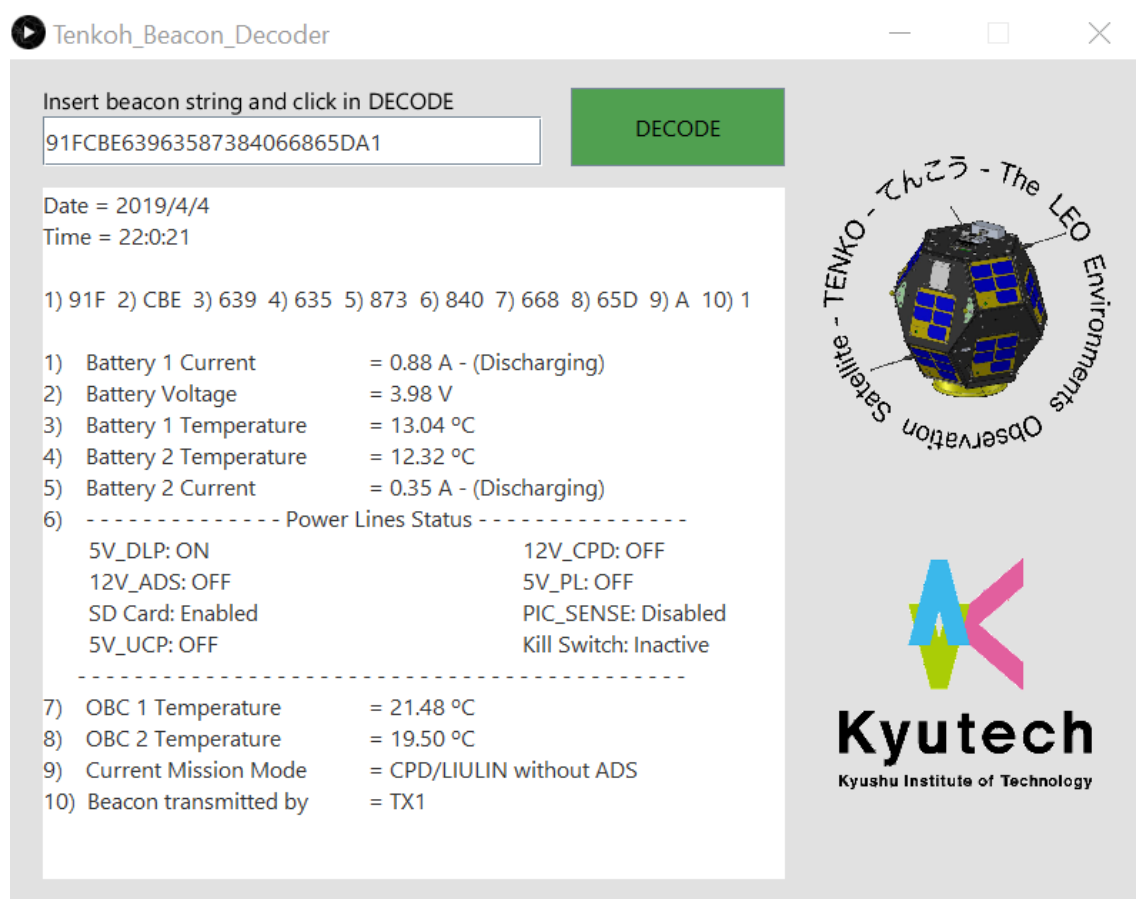


Figure 17. Last decoded beacon received from Ten-Koh over Greece on 19 March 2019. Note that “Date” and “Time” correspond to the time when the beacon was decoded, not when it was transmitted.

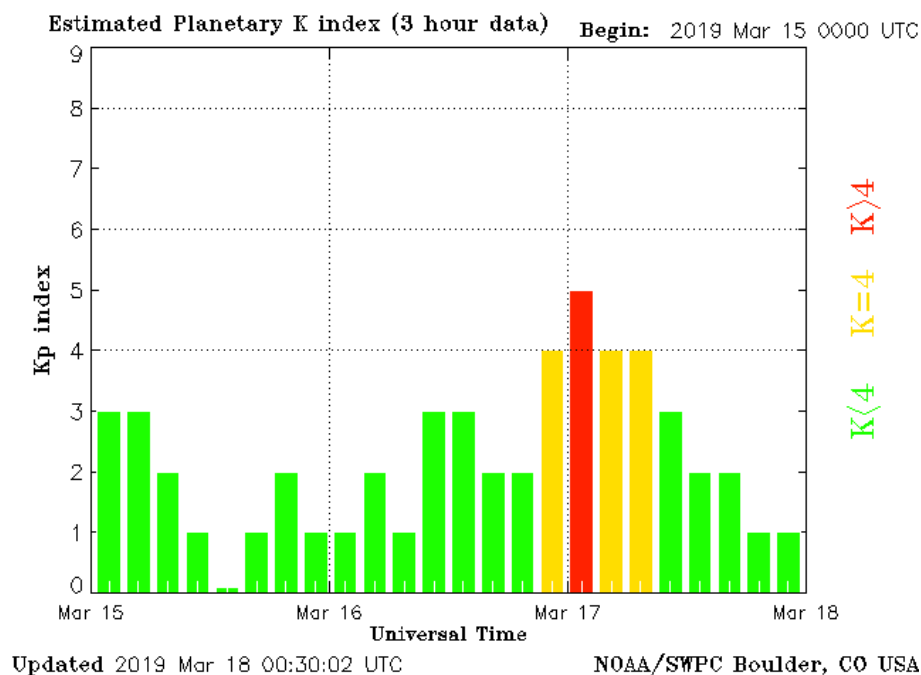


Figure 18. Estimated planetary index during the time the Ten-Koh anomaly occurred on 19 March 2019 (showed with the red bars).

At the time of writing, the satellite regularly transmits its beacon that includes correct on-board time and temperature data, and receives commands. However, conducting further experiments is currently impossible because the microcontroller governing the EPS is not reacting to commands or providing any telemetry. Because of the issues that the EPS controller has encountered previously, it is believed that the PIC's EEPROM is corrupted, which is preventing the reception of commands from the OBC via serial communication buses. The microcontroller reacts to hardware interrupts that are used to trigger a satellite reset, which confirms that the PIC itself is still functioning. Currently, the operations continue to try and recover full functionality of the spacecraft, which is plausible given the history of the previous failures.

5. Conclusions

The Ten-Koh satellite, launched on 29 October 2018, was the second satellite developed in a series of two, similarly-sized, low-cost and rapidly-developed small satellites from the Okuyama laboratory of the Kyushu Institute of Technology. The satellite was used to collect data pertaining to degradation of materials in space and high-energy particle fluxes in low-Earth orbit during the solar minimum.

Preliminary analysis of the data has shown that the main missions were able to acquire data within the expected ranges, except for the DLP that could not be operated due to the described power line failure. The CPD instrument has detected electrons, protons and galactic cosmic rays, which provides information about the radiation environment inside the spacecraft. In the case of the magnetometer, it has been shown that the selected design, where the instrument is attached directly on the top of the spacecraft without a boom, is able to produce measurements of the geofield that serve for attitude and science purposes. Above all, the in-orbit data have shown that the university-built satellite worked as intended, which is a success in its own right due to the limited resources that were available for its development.

The joint observations with the ARASE satellite show that the high-energy electrons mission on-board Ten-Koh has contributed to the field of particle science. Moreover, this mission has shown that small-satellites can be used as a complement to other missions investigating space environment.

A variety of failures, relating to both the satellite subsystems and the ground segment have been described. The most critical ones have been associated with triggering of over-current protection, degradation of EEPROM memory due to space radiation, and single-event upsets. These should be taken into account when developing future low-cost space missions, which will face similar challenges to those described in this paper. The best means of recovering from such failures has been found to be a power cycle of a unit, which should be implemented in future missions.

The abundance of radiation-related failures, with consequences as severe as disabling the entire satellite, indicates that understanding of the space environment and its effects remains vital. This shows that the mission objectives of Ten-Koh served a valid scientific purpose, and should be pursued by more missions. Space environment effects are especially important in the era where small satellites are proliferating because such spacecraft do not use radiation-hardened parts and thus are likely to suffer similar issues as the satellite described herein. Thus, the lessons learned described in this paper should be taken into account when developing similar satellites.

Lastly, the interplay between programmatic constraints and scientific return has been highlighted. One notable example of this is how discarding attitude control in order to meet the resources and schedule constraints has caused satellite rotation to be visible in CPD readings. Trading off cost, schedule, risk as well as return of the mission will be particularly important for small satellites, which cannot afford the luxury of postponing launches in order to meet their original scientific objectives. This is a large shortcoming of relying on piggyback launch opportunities, and space agencies worldwide should consider offering more flexible launch slots in order to maximize the return they get from small satellite missions. If the launch date cannot be postponed, spacecraft developers ought to consider reducing the scope of their mission by discarding certain objectives in order to satisfy the schedule constraint.

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Abbreviations

The following abbreviations are used in this manuscript:

ADC	Analog to Digital Converter
ADS	Attitude Determination Subsystem
COMM	Telecommunications subsystem of Ten-Koh
CCU	Communications Control Unit (of Ten-Koh COMM)
CFRP	Carbon Fiber Reinforced Polymer
CFRTP	Carbon Fiber-Reinforced Thermoplastic
ECU	Experiment Control Unit
EPS	Electrical Power Subsystem
GFRP	Glass Fiber Reinforced Polymer
GNSS	Global Navigation Satellite System
GOSAT	Greenhouse gases Observing SATellite
GPIO	General-Purpose Input Output
CPD	Charged Particle Detector, one of Ten-Koh payloads
COTS	Commercial off-the-shelf
DLP	Double Langmuir Probe, one of Ten-Koh payloads
I2C	Inter-Integrated Circuit (protocol)
JAXA	Japan Aerospace Exploration Agency
Kyutech	Kyushu Institute of Technology
LEO	Low Earth Orbit
LEOP	Launch and Early phase (of the Ten-Koh mission)
MCU	Main Controller Unit (of Ten-Koh OBC)
MGS	Main Ground Station
OBC	Onboard computer
OCP	Over-Current Protection
PCB	Printed Circuit Board
PIC	Programmable Integrated Circuit
PL	Payload
RTC	Real Time Clock
SPI	Serial peripheral interface (protocol)
TTL	Transistor–transistor logic
UHF	Ultra-High Frequency
UCP	Ultracapacitor
UV	Ultraviolet

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