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Operation Plan and Result of CubeSat Lunar Lander OMOTENASHI

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Abstract

This paper describes the nominal operation plan, results and future plans of OMOTENASHI, which was developed as the world's smallest lunar lander and launched as one of the sub-payloads of NASA's Artemis-1. OMOTENASHI is a 6U CubeSat designed with limited electric power resources and control capability compared with typical large spacecraft. Because OMOTENASHI will make its lunar landing directly from the lunar transfer orbit without going through a lunar orbit, it only takes about five days from rocket separation to landing. Thus, spacecraft power-on, critical operational events, and lunar landing operation must be conducted during this short period. Various solutions were considered for implementing the operation plan under such limited spacecraft capabilities and operation time. Unfortunately, the mission failed due to a malfunction immediately after rocket separation. Although telemetry could be received temporarily, the spacecraft was unable to control its attitude relative to the sun's direction and depleted its battery power. An open valve failure in the gas jet thruster was suspected to have caused the malfunction. After failing to make a lunar landing, OMOTENASHI is still traveling in space. But because the spacecraft revolves around the sun resulting in the sun's direction being gradually changed, it may be able to generate electricity in the near future. Search and recovery operations are being considered.

Keywords: CubeSat, lunar lander, operation plan, malfunction

Acronyms/Abbreviations

Attitude and Orbit Control System (AOSC)
Communication system (COM)
Chip-scale Atomic Clock (CSAC)
Delta Differential One-way Range (DDOR)
Deep Space Network (DSN)
Delta Velocity (DV)
Engineering Model (EM)
Flight Model (FM)
Generic Spacecraft Test and Operations Software (GSTOS)
House Keeping (HK)
Interim Cryogenic Propulsion Stage (ICPS)
Micro-Propulsion System (MiPS)
On Board Computer (OBC)
Orbiting Module (OM)
Outstanding MOon exploration TEchnologies demonstrated by NAno Semi-Hard Impactor (OMOTENASHI)
Power Control Unit (PCU)
Range and Range Rate (RARR)
Rocket Motor (RM)
Solar Array Panel (SAP)
Solar Array Simulator (SAS)
Spacecraft Information Base version 2 (SIB2)
Space Launch System (SLS)
Surface Probe (SP)
Trajectory Correction Maneuver (TCM)
Universal Time Coordinated (UTC)
flexible Adcs Cubesat Technology (XACT)

1. Introduction

Developed by JAXA, OMOTENASHI [1] was launched by SLS on 16 November 2022 as a secondary payload of NASA's Artemis-1 [2]. OMOTENASHI has two mission objectives. The first objective is to develop and demonstrate technologies for landing on the Moon with a CubeSat system. Using a low-cost, ultra-small spacecraft to land on the Moon would make future lunar exploration possible for universities, small and medium-sized companies, and even individuals. Landing on the Moon, which has no atmosphere, generally requires a large amount of propulsive force to decelerate the orbital velocity. However, it is difficult for a CubeSat-sized system equipped with liquid thrusters to achieve sufficient deceleration for a lunar landing. OMOTENASHI uses an ultra-small solid rocket motor that cannot control the propulsive force. And given its size and mass, a radar system to measure velocity to the lunar surface cannot be mounted. Designed for a semi-hard landing instead of a soft landing, OMOTENASHI has remaining velocity of less than 50 m/s at landing and thus requires shock-absorbing technology to survive the landing impact. OMOTENASHI also requires precise orbit determination because its solid rocket motor can only adjust the direction of deceleration and ignition timing. The second mission objective is to measure the radiation environment outside Earth's magnetosphere. The shielding effect of Earth's magnetosphere results in low levels of cosmic radiation on the ISS. It is thus necessary to know such values in view of a more critical radiation environment expected on the trajectory to the Moon and Mars. The radiation environment outside Earth's magnetosphere has rarely been measured. OMOTENASHI has an ultra-compact dosimeter weighing several tens of grams. Although the ultra-small system cannot make high-precision observations, mounting it on many spacecraft in the future would make it possible to collect observation data at many points and times.

In order to accomplish these missions with a 6U-size CubeSat, each component must be ultra-small. OMOTENASHI adopted the components for CubeSats, such as the attitude control unit and gas jet thrusters. The solid rocket motor, which is used for deceleration during a lunar landing, and the telecommunication system were newly developed to fit the size of OMOTENASHI. The main computer was a commercial off-the-shelf microcomputer, which had already been demonstrated in orbit. Compared to large spacecraft, the system design was extremely lacking in terms of electric power resources, control capability, and computing resources. The operation had to be accomplished using the limited resources available. In particular, the stringent power requirements caused such operational problems as the inability to activate multiple high-power components simultaneously. Moreover, OMOTENASHI's orbit control capability made insertion into a lunar orbit difficult. The spacecraft was intended to land on the Moon directly from a lunar transfer orbit without going through a lunar orbit [3,4]. In the direct landing orbit, it only takes about five days from rocket separation to landing. Therefore, many critical operations such as power-on, initial checkout of onboard components, orbit control, orbit determination and lunar landing operation must be done during this short period.

In order to conduct the operations within the spacecraft's capabilities and short operation period, various solutions were considered in terms of spacecraft design and operation procedure. For example, an automatic telemetry check function was considered for the initial checkout to identify problems. The manual thrusting operation for initial thruster maintenance to stabilize the force posed the risk of causing a deviation from the allowable angular momentum of the reaction wheel. The thrusting parameters are directly calculated from current telemetry on the operation control system. Orbit maneuvers ranging from a swing-by orbit to a direct landing orbit were crucial for the mission, but these operations consume much time and high power. Thus, a strategy for orbit maneuvers was intensively studied.

Training is just as important as operational planning. In order to realize meaningful training, an operation simulator that behaves like the flight component was developed and connected to the actual operation control system. The operation simulator has the same onboard computer and telecommunication system as the FM. The software is also the same as that implemented in the FM. It is difficult to gage the behavior of attitude sensors, reaction wheels, and gas jet thrusters in a laboratory. A real-time simulator was thus developed to calculate the behavior and attitude dynamics of attitude components while connected to the onboard computer of the operation simulator instead of real components.

Unfortunately, the mission failed due to a malfunction immediately after rocket separation. Telemetry could not be demodulated even at the scheduled signal acquisition time, and an emergency operation was conducted. Although its telemetry could be received temporarily, the spacecraft spun at over the allowance angular velocity, and the solar panels headed in the direction opposite of the sun. The spacecraft was unable to generate electricity, resulting in the loss of radio signals due to depleted battery power. An analysis of the partially acquired telemetry and radio signal strength history showed that an open valve failure in the gas jet thruster caused the malfunction.

2. System

2.1. Spacecraft system

2.1.1. Hardware

OMOTENASHI is a 6U-sized CubeSat with a mass of approximately 14 kg. It consists of three modules: Orbiting Module (OM), Surface Probe (SP), and Rocket Motor (RM). Figure 1 shows the appearance of OMOTENASHI.

The OM has basic bus functions such as attitude control, orbit control, data processing, communications and power supply. All commands and telemetries are interpreted or generated in the OM main computer. Immediately before a lunar landing, the OM separates from the SP+RM and completes its role. In order to fit each component in a limited space, OMOTENASHI employed small products for CubeSat. Some components were newly developed as necessary. For example, JAXA developed the thin-film type solar array and demonstrated it twice in orbit before. This is mounted on the PY plane of the OM. The batteries are three series of consumer 18650-type cells. The communication system (COM) transmits or receives X-band radio for commands and telemetries, as well as for RARR and DDOR for orbit determination. COM deals with UHF-band radio for the campaign of amateur radio as well. COM was newly developed as an ultra-compact type of integrated component with a power control unit (PCU). The main computer is a consumer SH2 microcontroller. Moreover, the memory, DCDC converter, communication driver, crystal oscillator, real-time clock and MEMS IMU are all implemented on the same microcontroller board. The OM has a chip-scale atomic clock (CSAC) that generates a precision counter for communication frequencies and spacecraft time. The OM mounts XACT, which is an attitude control unit for CubeSat. XACT has four sun sensors, a star tracker, three reaction wheels, and an attitude calculation computer. The three-axis pointing accuracy is 0.01 deg. The OM has a Micro-Propulsion System (MiPS) equipped with gas jet thrusters that use R-236fa as propellant gas. The thrust force is 25 mN. The MiPS has four thrusters, and two MiPS are mounted on each side of the spacecraft. Figure 3 shows the directions of all eight thrusters. A1-A4 are axial thrusters for orbit maneuvers; T1-T4 are tangential thrusters for spin-up operation. Figure 4 shows that the internal structure of MiPS consists of a tank, plenum, internal valves, thruster valves and nozzles. The OM also carries a camera, an ignition system for the RM [5], a separation mechanism for the SP+RM, and a radiation monitor to address one of the mission objectives. The OM has severe power constraints. For instance, if the OM drives XACT and MiPS simultaneously in high-power mode, power consumption will exceed the PCU's capability. Current limiters are not implemented in the PCU, however, in order to avoid power outages during landing operation. Therefore, attitude maneuver and orbit maneuver operations require that software design and operation plans do not deviate from the power limits.

The SP is a lunar landing probe attached to the RM. The success or failure of the OMOTENASHI mission in making a lunar landing will be determined by acquiring radio signals transmitted by the SP before and after landing. The SP has a primary battery, an ultra-compact communicator, a microcomputer and a deployable antenna as minimum functions. The SP is designed to land on the lunar surface with semi-hard impact at a speed of 50 m/s. Crushable material is installed to reduce the impact [6]. The electrical parts are filled with epoxy to withstand 8500G.

The RM is a solid rocket motor used in deceleration for landing. The RM burns for about 17 seconds with 470 N to decelerate the SP+RM by 2500 m/s.

Table 1. Specifications of the OM

Mass	8500 g
Main structure	Aluminum
Power system	Solar cell (Body mount) 8s3p 23.8W Li-Ion battery 3s1p 38.4 Wh Peak power tracking for solar cells Battery charging/discharging control
Telecommunication	X-band uplink/downlink, RARR, DDOR UHF-band uplink/downlink
Attitude control	Three-axis Pointing accuracy 0.01 deg Spin up to 300 rpm
Propulsion	Total impulse 560 Ns for two modules Thrust 25 mN x 4 for each module Propellant R-236fa (Isp 40 sec)
Onboard computer	Telemetry and command handling Timeline commands, Data recorder
Radiation monitor	Semiconductor detector Two sensors: for Proton and GCR

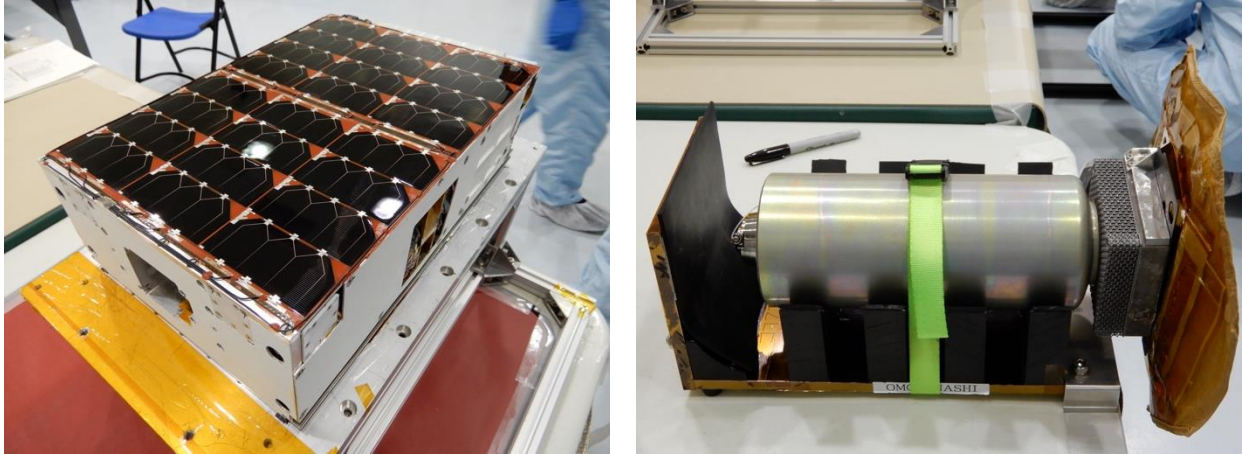


Fig. 1. Appearance of the spacecraft (left: OM, right: RM+SP)

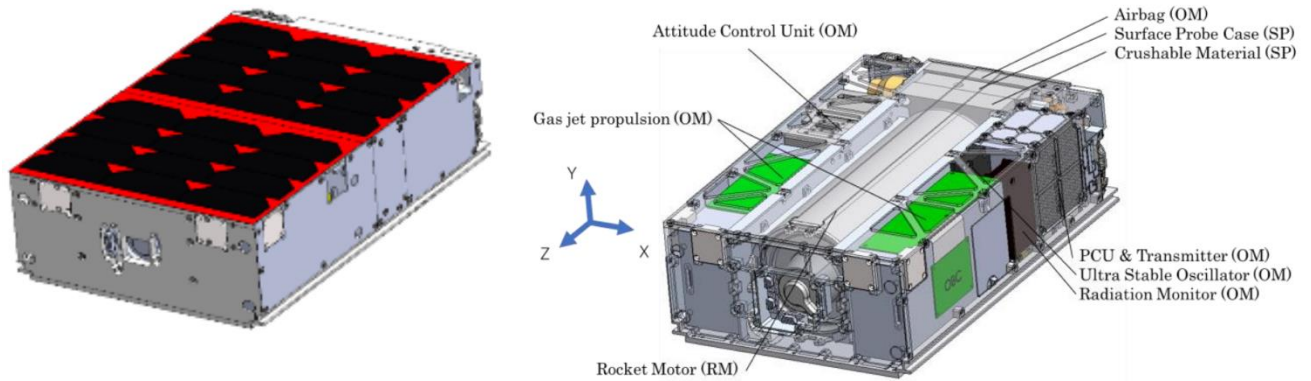


Fig. 2. CG model of the spacecraft

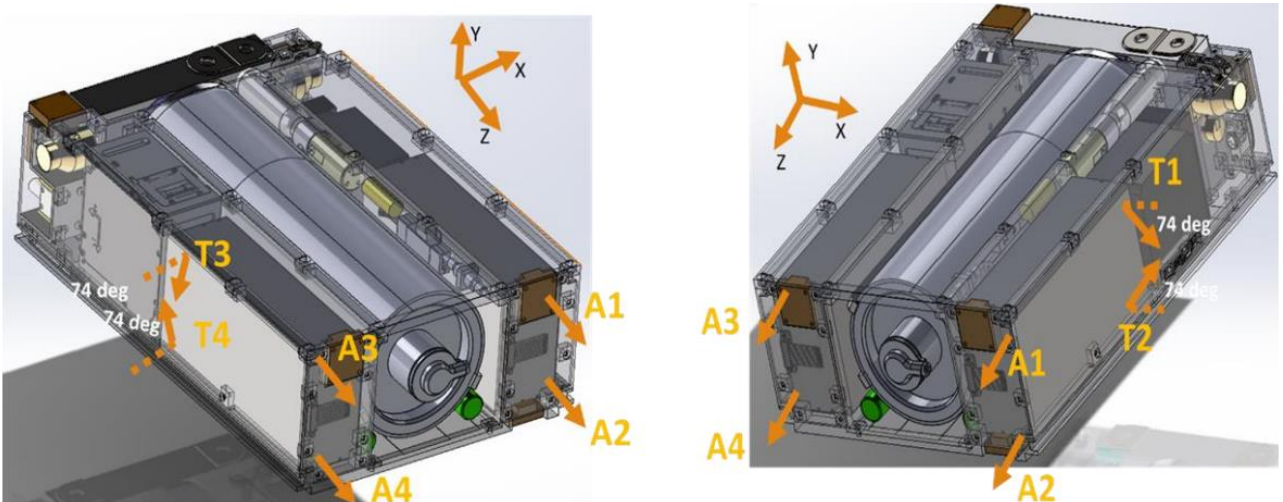


Fig. 3. Thruster allocation of MiPS

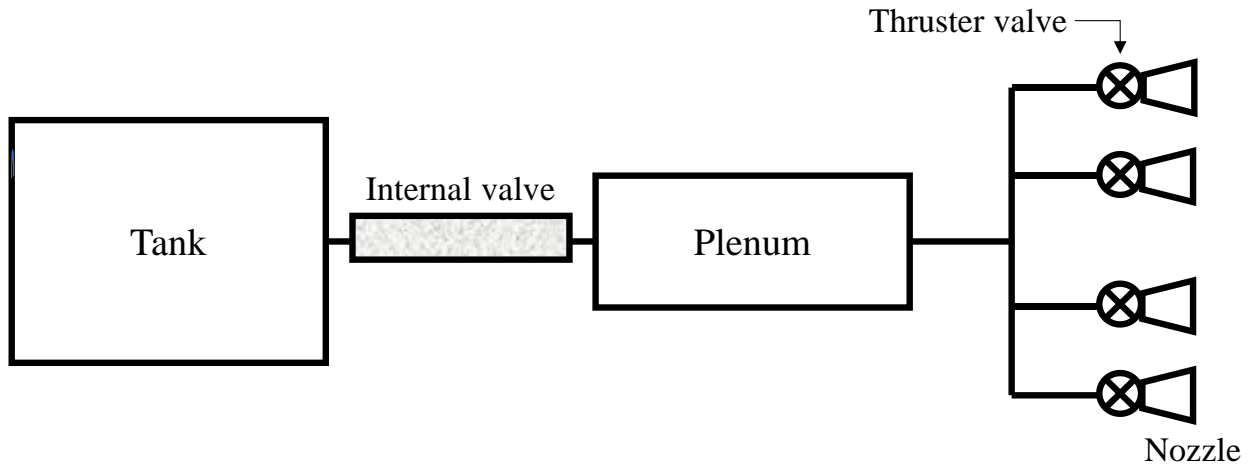


Fig. 4. Internal structure of MiPS

2.1.2. Software

The OMOTENASHI software was developed using μ ITRON, a well-known real-time OS. A functional unit administrated by the OS is called a task, and multiple tasks are processed in parallel. Software routines and functions related to a certain function are implemented as a single task (e.g., AOCS Task, PCU Task, Database Task). Figure 5 shows that the software architecture is defined as a tree of tasks. Data interfaces between the tasks are definitely managed by the Main Task to avoid spaghetti code. Commands and telemetries of OMOTENASHI are compliant with the CCSDS protocol. Typical software functions are described below.

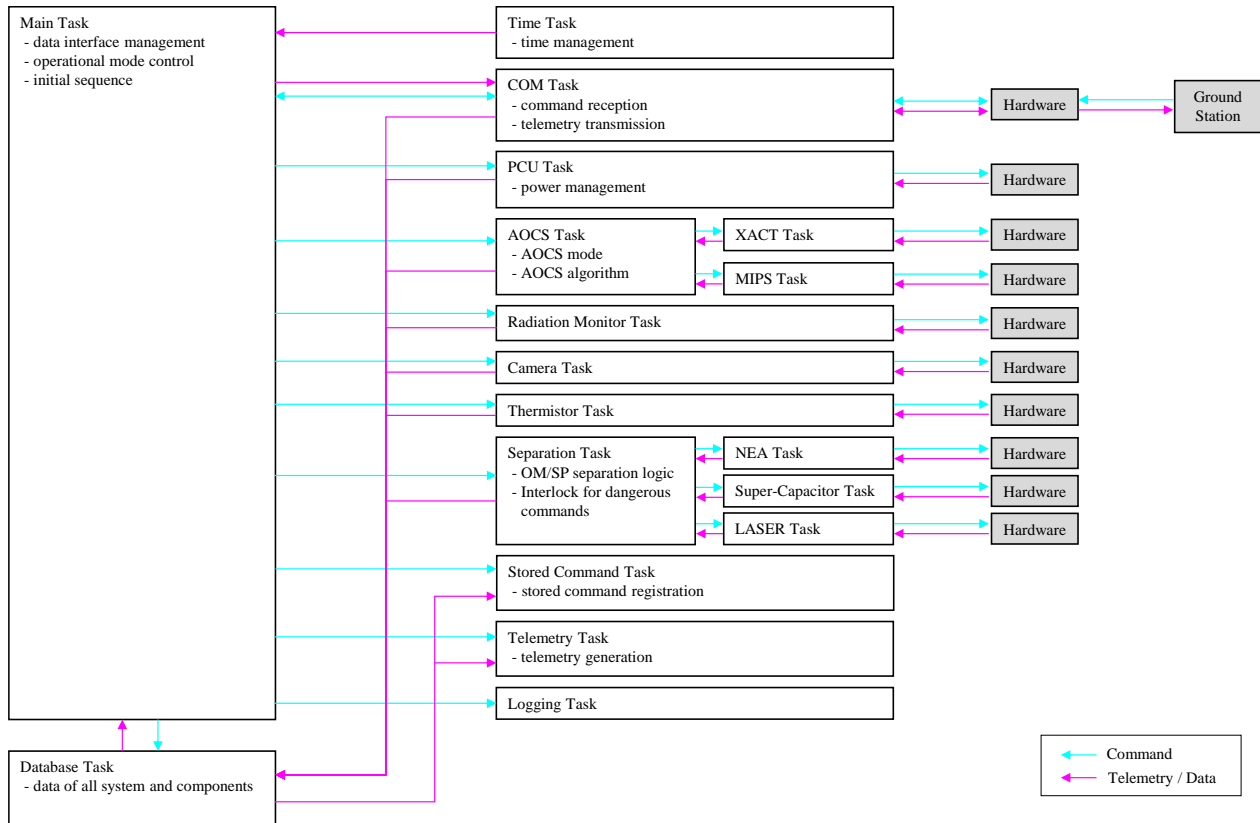


Fig. 5. Software architecture and data flow

OMOTENASHI has three types of commands: real-time commands, stored commands, and macro commands. Real-time commands are executed immediately upon receipt by the spacecraft’s OBC. Stored commands are sent with a future execution time, and are executed at the specified time. Macro commands are sent with the relative time, and are executed after the relative time from the start command of the macro sequence.

The types of telemetry include HK telemetry, status telemetry, time calibration telemetry, and fill data telemetry. HK telemetry, the basic status of the current spacecraft, is used for daily checks and generated every second. Status telemetry includes the detailed data not included in HK telemetry. A request command specifying the status number is needed to obtain status telemetry. Time calibration telemetry is transmitted to synchronize between a spacecraft time indicator (TI) and UTC as accurately as possible; HK telemetry also contains TI, but the time is uncertain when the TI is obtained during the HK telemetry cycle. Time calibration telemetry has TI obtained immediately before the telemetry transmission. Fill data telemetry is transmitted if the telemetry buffer has no data.

The OMOTENASHI software has a test mode function. The status is ORBIT mode or TEST mode. If the ground test jig input a voltage of 3.3V to the digital input port, the software is started in TEST mode to avoid turning on MiPS. The software is started in ORBIT mode in space, and all AOCs functions are activated. ORBIT mode or TEST mode can be changed by operation command as well as voltage input.

The AOCs functions have five modes; normal control mode, rate damping mode, unloading mode, spin-up mode, and orbital control mode. XACT of the attitude control unit can independently perform sun acquisition and three-axis attitude control. Thus, there is no need to implement these functions as OBC software. The attitude is controlled only by XACT in normal control mode. If MiPS is used for external force control, the software’s attitude control task must properly operate XACT and MiPS, considering that both cannot be used simultaneously with high-power utilization due to the severe electric power condition of OMOTENASHI. In rate damping mode, when the spacecraft is rotating more than the allowable angular momentum of the reaction wheels, the spacecraft is decelerated by injecting the gas propellant of MiPS in the direction opposite that of the rotation. If the spacecraft’s angular velocity exceeds a certain threshold at separation from the rocket, rate dumping is automatically initiated. Unloading mode specifies the target rotation speed of the reaction wheel (e.g., 0 rpm) and injects the gas propellant of MiPS to approach the target speed. Spin-up mode injects MiPS tangentially for a long time to make the spacecraft spin at high speed in the Z-axis direction. Orbital control mode is used for orbit control in DV1 and TCM. The DV operation is divided into phases due to power supply limitations. Both attitude correction by XACT and orbit change by MiPS are conducted in another phase. And since the battery power reaches a low level in DV operation for 10 to 15 minutes, the charging phase and discharging phase are separated. Orbit control mode, angular momentum correction mode, attitude correction mode, and battery charging or discharging are performed according to the diagram shown in Figure 6.

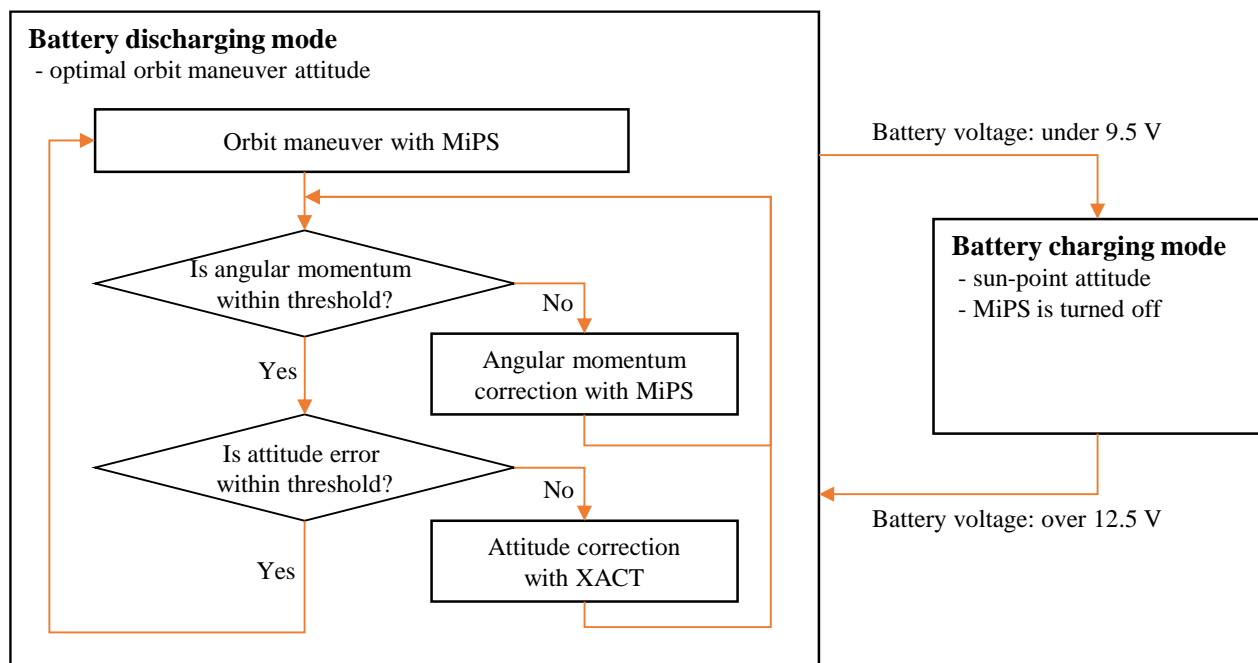


Fig. 6. Flow of orbital control mode

connect the EM battery. The communication unit of the operation simulator is connected to the actual ground equipment via coaxial cable, enabling it to transmit commands and receive telemetries. Therefore, it is possible to conduct realistic operational training using the same operation room and devices.

3. Nominal operation plan

3.1. Mission scenario

After flying out from the dispenser, OMOTENASHI automatically initiates the attitude acquisition sequence in order to face body-mounted solar cells towards the sun. After the first health check of the components, three-axis attitude control is started using the star tracker and reaction wheels of XACT. The initial maintenance of MiPS is conducted before MiPS is used for an orbit maneuver. The orbit maneuver to enter into lunar impact orbit (DV1) is performed by cold gas jet propulsion. DV1 will be about 10 m/s. Trajectory correction maneuver (TCM) could also be performed if necessary. About 45 minutes prior to lunar impact, the spacecraft starts the landing preparation sequence that includes an attitude maneuver and spinning up for RM firing. DV2 is conducted by the RM and its deceleration will be 2500 m/s. Soon after ignition of the rocket motor, the OM is separated. After DV2 is completed, the SP is deployed with semi-hard impact made with the lunar surface. The SP is designed to survive for a few minutes on the lunar surface. Figure 8 shows the mission scenario. Ultra-small radiation monitors are mounted on the OM to measure the radiation environment of the trans-lunar orbit and lunar surface.

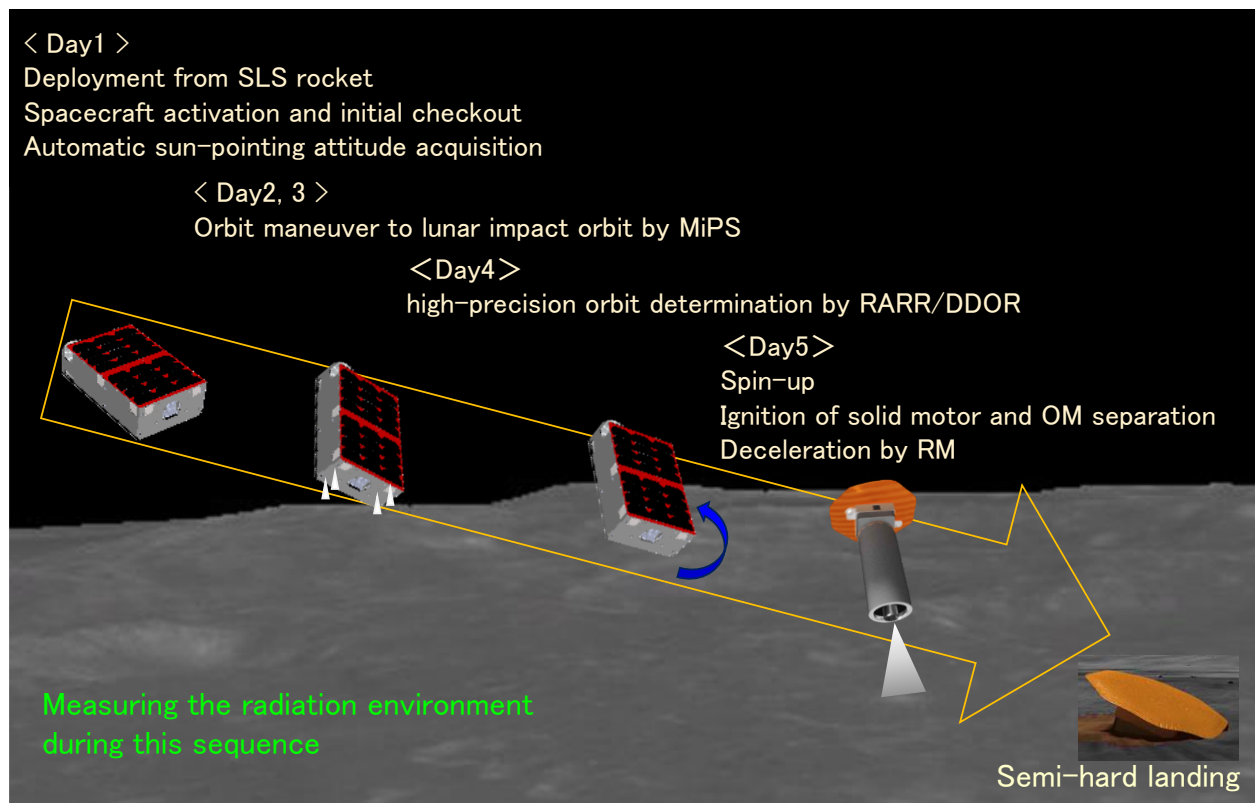


Fig. 8. Mission scenario

3.2. Operation events

3.2.1. Initial sequence of spacecraft

When the spacecraft is released from the dispenser, COM and the OBC are turned on. In order to prevent the OBC from operating near the rocket for safety design reasons, it is designed to start operating 30 seconds after the power line is connected. This function is achieved by a capacitor mounted on the OBC.

Once the OBC starts operating, each task is activated by the μ TRON process to begin multitasking. Then the initial sequence is executed. The initial sequence turns on the components, sets basic parameters, starts downlinking HK telemetry and sun acquisition control by XACT, and performs rate dumping by MiPS if necessary. Table 2 shows the contents of the initial sequence.

Table 2. Initial sequence

1	Set test mode or on-orbit mode in database
2	Turn on radiation monitoring
3	Start LOG
4	Set COM to 2-way mode
5	Turn on CSAC VCC of COM
6	Start synchronizing the OBC counter with CSAC
7	Set reference frequency of COM to CSAC
8	Set CMD bit rate of COM to 125bps
9	Set TLM bit rate of COM to 4096bps
10	Set TLM modulation of COM to 1.2 rad
11	Set COM's ranging modulation to 0.3 rad
12	COM's transmit antenna set to ANT2
13	Transmit output of COM is set to Low Power
14	Ranging of COM is turned ON
15	Tx of COM is turned ON
16	Start transmission of HK telemetry (ID1, period 1s)
17	XACT ON
18	Set rotation during XACT solar pointing (0.5 deg/s)
19	Start XACT telemetry
20	Start AOCS rate dump (in on-orbit mode and when exceeding RW allowable angular momentum)
21	Transition to main process

After the initial sequence, the spacecraft starts a slow spin of 0.5 deg/s around the PY axis, while maintaining the sun point. The spacecraft has two downlink antennas on the PZ and MZ planes. However, the two antennas cannot transmit simultaneously, and one has hemisphere coverage. The slow spin is necessary to ensure reception of the telemetry at the ground station.

Given the uncertainty of the ground station being assigned at the time of rocket separation, the onboard software automatically executes the initial sequence. The spacecraft maintains flight with sun-point attitude and charges its battery until receiving a command.

3.2.2. Initial checkout

As soon as the ground station receives telemetries, an initial checkout is conducted by checking the periodically downlinked HK telemetry and the status telemetry obtained by command. Both kinds of telemetry are checked using three methods. The first is visual judgment by the operator. Each normal telemetry on GSTOS is displayed in green but changes to yellow or red if it exceeds the predefined threshold of alert level or anomaly level. The operator should check the telemetry window immediately after acquiring telemetry. Second, the GSTOS telemetry check function is used to identify all acquired telemetry as normal or abnormal. This prevents the operator from missing problems. Third, the experts in charge of each subsystem or component check the telemetry. As the pre-considered thresholds cannot necessarily diagnose any anomalies in orbit, the experts make a comprehensive judgment based on the current status and time trends.

3.2.3. Three-axis attitude control

After the initial checkout is completed, three-axis attitude control is initiated. First, a check is made to see if attitude determination by the star tracker required for three-axis attitude control has been successful. Then commands issued from the ground input the orbit and time information necessary for XACT processing. As the input information varies depending on the launch date and orbit, an operation procedure corresponding to the launch date is prepared, and the operation simulator checks the soundness of the procedure. Next, XACT control mode is changed to three-axis control mode, and when the target quaternion is commanded, the spacecraft maneuvers to an arbitrary attitude. The steady-state posture of OMOTENASHI is set with the solar panels facing the sun and the downlink antenna attached to the MZ pointing toward Earth as much as possible. The downlink antenna's pointing angle is determined by the geometric position of the Sun, Earth and the spacecraft, and also affects the telemetry bit rate that can be communicated.

3.2.4. Initial maintenance of thruster

Subsection 2.1.1 already described the internal structure of MiPS. The results of pre-launch ground tests revealed that some liquid propellant contained in the tank had moved to the plenum due to a small leak in the internal valve. Although the small internal leak was deemed acceptable during the five-day mission, liquid propellant in the plenum area caused thrust instability. Therefore, the liquid propellant of the plenum was to be released before full-scale thruster operation.

The release of liquid propellant was to be performed simultaneously from two tangential thrusters, and thrusting was to be performed so that the spacecraft’s angular momentum would not deviate from the allowable value of the reaction wheels. If the angular momentum approaches the upper limit, unloading will be performed by manual operation. The automatic unloading function implemented in the onboard software is designed by assuming stable thrust force. However, performing this function before the release operation is dangerous. In manual unloading, the thruster valve to be driven and the seconds of injection should be calculated based on the current telemetry. The injection seconds are short enough to be safe, considering the instability of the thrust. Manual unloading is thus repeated multiple times to reduce the accumulated angular momentum.

As the manual unloading parameters are calculated based on telemetry values, it is difficult to prepare commands in advance. This operation has a high risk of operational errors due to human mistakes. In order to eliminate operational errors, command parameters are automatically calculated in GSTOS from the current telemetry. The calculation results are then copied and pasted into the command window, instead of being input manually. Operational training was conducted sufficiently.

3.2.5. Preparation of DVI

The DVI sequence is critical to mission success and DVI orbit planning must consider the orbit determination, thrust force, and power supply environment. The DVI sequence is performed in discharge mode and charge mode (described in Subsection 2.1.2). The orbit maneuver is conducted in discharge mode.

The time of each battery mode is an important factor in DVI orbit planning. However, it varies depending on the attitude and SAP power generation because the thrusting attitude does not always point the SAP toward the sun. In addition, the amount of DV in the translational direction varies due to the attitude disturbance caused by the thrust imbalance between MiPS located on the left and right sides. Test DVI operation is conducted before the DVI operation. The time required for the discharge and charge modes, and the on-orbit experimental values of the attitude disturbance are obtained from the Test DVI results and reflected in the DVI parameters.

DVI orbit planning uses the orbit determination from RARR in addition to the separation orbit information from the launch vehicle, and calculates parameters such as the total thrusting time, thrusting attitude quaternion, and charging attitude quaternion based on the Test DVI results and orbit determination. A DVI start command is then created with the parameters. To avoid human mistakes, the created command is verified with an operational simulator. When the verification is satisfactory, the command is sent to the spacecraft and the DVI function of the onboard software is executed.

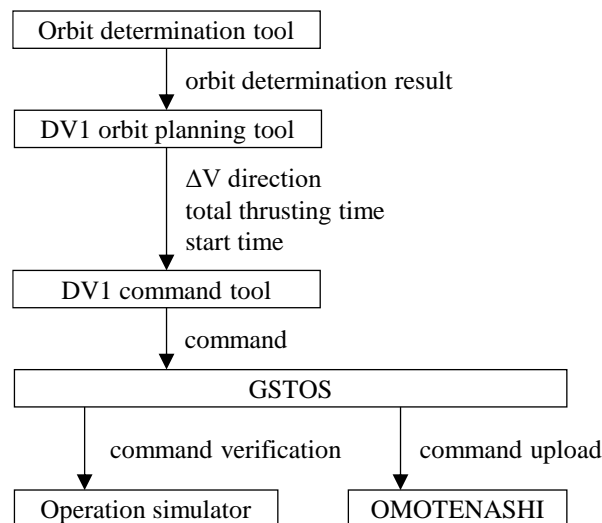


Fig. 9. Preparation procedure of DVI

3.2.6. DV1

DV1 repeats the discharge and charge modes, taking about 8 to 26 hours. Such minimum operations as a telemetry check are only performed during a command and telemetry operation. After that, radio transmission is stopped to save battery power. TCM is performed in almost the same strategy as DV1.

3.2.7. Preparation of DV2

In deceleration using the RM, orbital velocity against the ground will be below 50 m/s. The only controllable parameters are the RM ignition time and attitude. The ignition time, in particular, greatly affects the altitude at the end of the RM burn. If ignition occurs earlier than the optimal time, the RM will free fall from a very high altitude after RM combustion is completed. If ignition is delayed, the vehicle will impact the lunar surface before deceleration by the RM is completed. Therefore, after the completion of DV1 and TCM, precise orbit determination will be performed by RARR and DDOR (using JAXA station UDSC64 and NASA DSN stations at Goldstone and Canberra) to accurately determine the spacecraft position from triangulation with baseline lengths to the east, west, south, and north of Earth. Gas jet thrusting operations using MiPS cannot be conducted after DDOR because they would adversely affect orbit determination accuracy. The time calibration operation is conducted using the time calibration system. The RM burn time also depends on temperature. The DV2 parameter tool calculates the DV direction and ignition time from the precise orbit determination and the predicted RM temperature. N. Morishita *et al.* [9] explained the strategy behind the DV2 parameters. The DV2 parameters are converted by the DV2 command tool into a command format that can be interpreted by the OBC. Once created, the commands are verified with the operation simulator. In terms of planetary protection [10], the probability of Mars impact must be analyzed to confirm that the probability of such impact over the next 50 years is less than 10^{-4} , even if the RM is ignited at any time and in any direction due to a malfunction after uplinking the DV2 command sequence. When command verification by the operational simulator and the probability of Mars impact are both satisfactory, the commands are sent as stored commands. The spacecraft then waits until the registered time for execution of the commands.

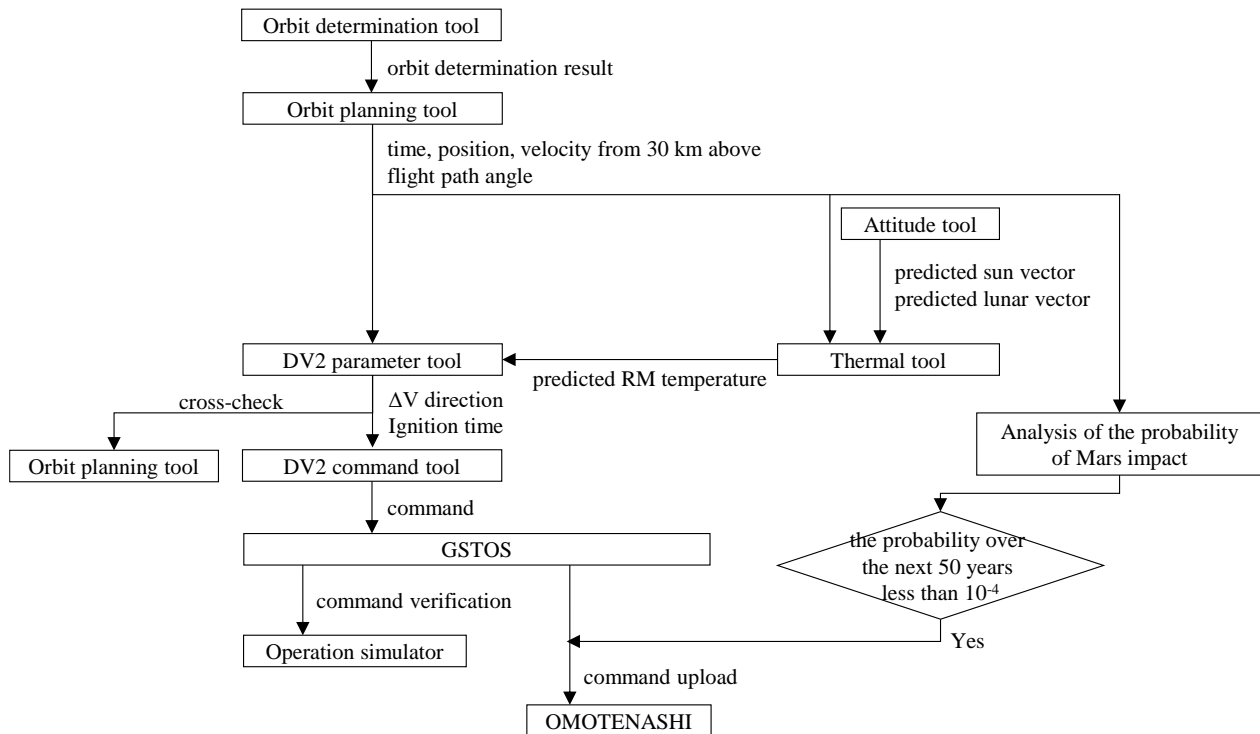


Fig. 10. Preparation procedure of DV2

3.2.8. DV2

The DV2 sequence is executed entirely by stored commands. First, a dummy command is executed. The dummy command does not have a specific action, but the stored command execution counter is increased, showing that the DV2 sequence started at the specified time. The unloading of the reaction wheels is performed next, with the RW rotation speed of the X and Y axes being set to 0 rpm and that of the Z axis to 4000 rpm. This unloading operation prevents attitude disturbance of the X and Y axes, and provides an initial speed for Z axis rotation when the reaction wheels are stopped immediately before spin-up. Next, MiPS plenum pressure is set for spin-up thrusting. COM sets the downlink power to high power and the telemetry bitrate to 256 bps. The antenna is pointing out of the earth direction during spin-up; thus, a high-power downlink and a low telemetry bitrate make the acquisition of telemetry easy. Next, hazardous commands (defined as the SP power command, OM and SP separation command, and laser ignition command) must be unlocked. The SP is then turned on for 40 seconds to test its condition. Since the SP has a primary battery, it is turned on only for the short time needed to complete the check. Next, the attitude maneuver for the RM burning direction is performed. After this maneuver, the SAP is out of the attitude of facing the sun. SAP power generation decreases. XACT is then turned off, with the angular momentum of the Z-axis reaction wheel being released and the spacecraft beginning to rotate. Next, MiPS is injected tangentially for a total of 684 seconds to spin up the spacecraft to approximately 5 Hz, which means that the spin-up stabilized the attitude during the RM burn. After the spin-up is completed, the SP is turned on 20 s before RM ignition. At 41 ms before RM ignition, the pin puller is driven to uncouple the OM and the SP+RM. Finally, RM ignition separates the OM and SP, and the SP decelerates approximately 2500 m/s to land on the lunar surface.

Table 3. DV2 sequence

ignition time - 0:45:00	Dummy Command
ignition time - 0:42:40	X-axis unloading
ignition time - 0:40:40	Y-axis unloading
ignition time - 0:38:40	Z-axis unloading
ignition time - 0:36:40	X-axis unloading
ignition time - 0:34:40	Y-axis unloading
ignition time - 0:32:40	Z-axis unloading
ignition time - 0:30:40	X-axis unloading
ignition time - 0:28:40	Y-axis unloading
ignition time - 0:26:40	Z-axis unloading
ignition time - 0:24:50	MiPS plenum pressure setting
ignition time - 0:24:43	Status telemetry downlink
ignition time - 0:24:40	COM high power mode shift
ignition time - 0:24:20	Unlock hazardous commands
ignition time - 0:23:40	SP ON for test
ignition time - 0:23:00	SP OFF for test
ignition time - 0:22:40	COM TLM bit rate 256 bps change
ignition time - 0:22:20	DV2 posture maneuver
ignition time - 0:15:40	XACT RW Power OFF
ignition time - 0:15:20	XACT OFF
ignition time - 0:15:00	Spin-up 1st (590 sec)
ignition time - 0:03:10	Spin-up 2nd (94 sec)
ignition time - 0:00:20	SP ON
ignition time - 0:00:10	MiPS OFF
ignition time - 0:00:00.41	PIMPLER ON
ignition time - 0:00:00.00	Laser Ignition

3.2.9. SP operation

The SP turns on 20 seconds before RM ignition, and the SP radio can be received at the ground station. Thereafter, the carrier frequency of the radio is designed to change according to the impact measured with the accelerometer. By monitoring the carrier frequency, the timing of lunar arrival can be determined. If the SP lands on the Moon with low impact, the lunar landing is determined via radio confirmation after the scheduled landing time. Given the primary battery's capacity and the severe thermal interface with the ground, the SP can only survive on the lunar surface for several minutes.

4. Operation results

4.1. Emergency operation with critical trouble

OMOTENASHI was launched from NASA's Kennedy Space Center on 16 November 2022 at 15:47:44 UTC. OMOTENASHI was separated from the SLS launch vehicle at approximately 19:30 UTC. The first operation used NASA's DSN Madrid station DSS53. Provided that the initial sequence proceeded normally, the completion of sun acquisition was assumed to take 10 minutes at most. After that, the spacecraft was set to rotate in 12-minute cycles. As the spacecraft repeated a 6-minute receivable / 6-minute non-receivable condition, radio waves were expected to be received at the ground station after a maximum of 22 minutes. However, telemetry could not be received at the expected time of 19:52 UTC. The NASA DSN station informed the OMOTENASHI team about weak signal strength and the spacecraft's high-speed rotation. It was thus difficult to receive radio waves. As an emergency operation, an uplink from the ground station was started, the downlink of the spacecraft was set to high power mode, and then the telemetry was received and demodulated. Figure 11 shows the telemetry at that time. Confirming the telemetry, the solar cell (PY plane of the spacecraft) was facing opposite to the sun, and the spacecraft was rotating at about 80 deg/s around the Y axis. This rotation speed exceeded the allowable angular momentum of the reaction wheels and sun acquisition control by XACT could not be derived. It took about 10 minutes to understand the situation and discuss the solution. A decision was made to re-execute rate dumping using a command to reduce the spacecraft's rotational speed. During the rate dumping, the downlink was turned off to conserve battery power. However, the rate dumping was not completed until battery depletion, and spacecraft power was assumed to be turned off. Since then, there has been no communication with the spacecraft. Unfortunately, the failure occurred shortly after the start of the operation, thus making it impossible to conduct the nominal operation plan for the rest of the day.

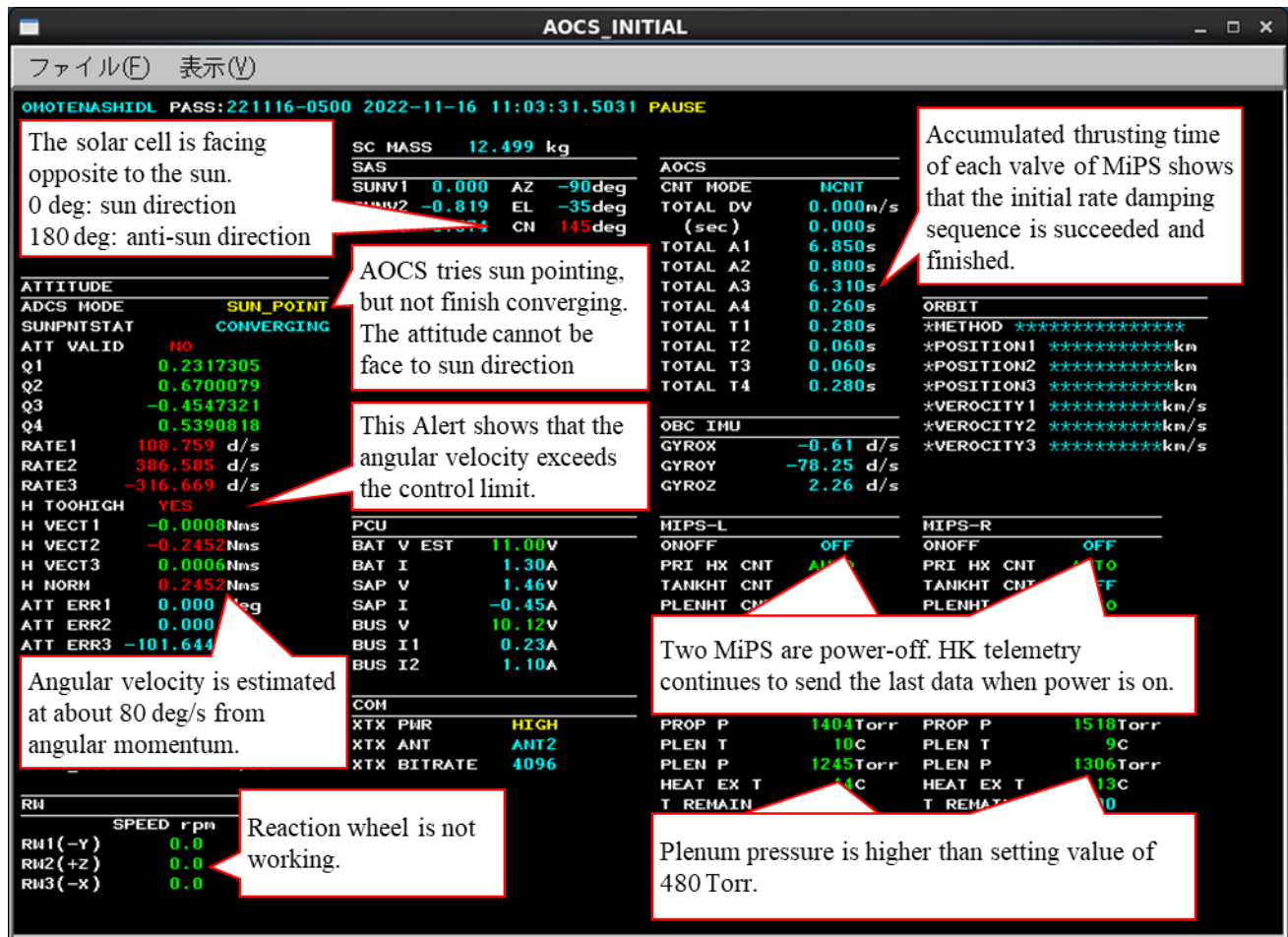


Fig. 11. The first acquired telemetry

4.2. Failure scenario

After the operation, NASA provided reception signal to noise ratio data recorded at the DSN station. Figure 12 and Figure 13 shows the reception S/N graph. According to this graph, the signal strength is oscillating. OMOTENASHI's downlink is transmitted from the antenna mounted on either the PZ or MZ plane. The default setting is the MZ plane. Spacecraft rotation causes the signal strength oscillation because the antenna's pointing angle to the earth changes according to the rotation angle. Analyzing the S/N data makes it possible to estimate the spacecraft status during periods of no telemetry reception. Based on the telemetry data and the reception S/N data, the failure scenario shown in Table 4 is considered.

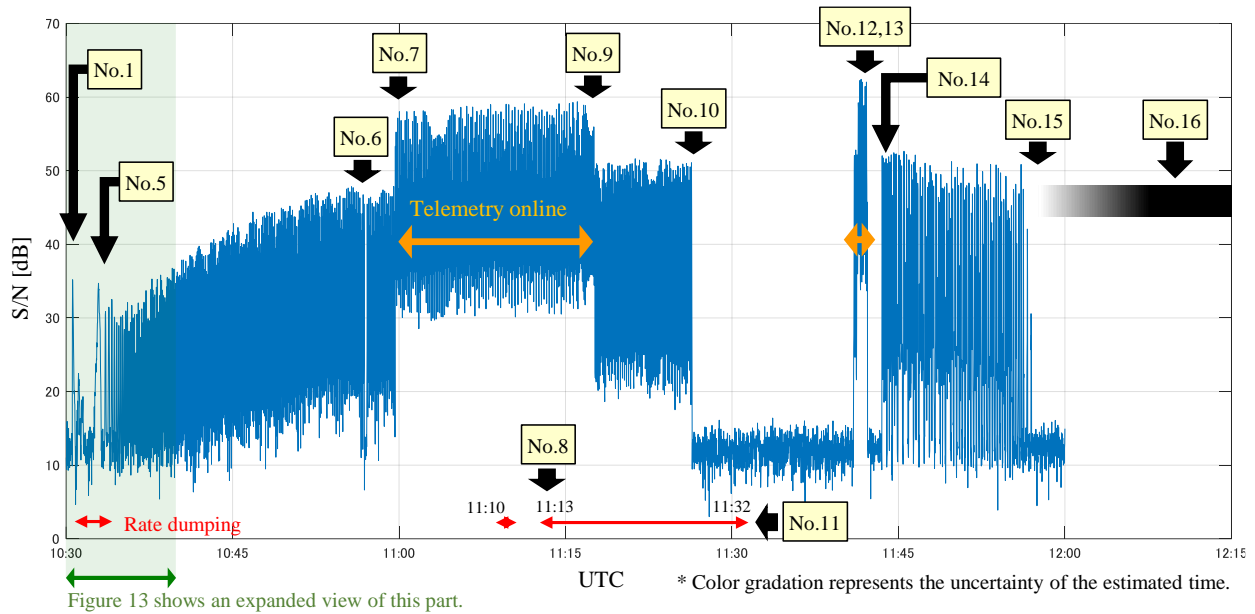


Fig. 12. Reception Signal to Noise Ratio

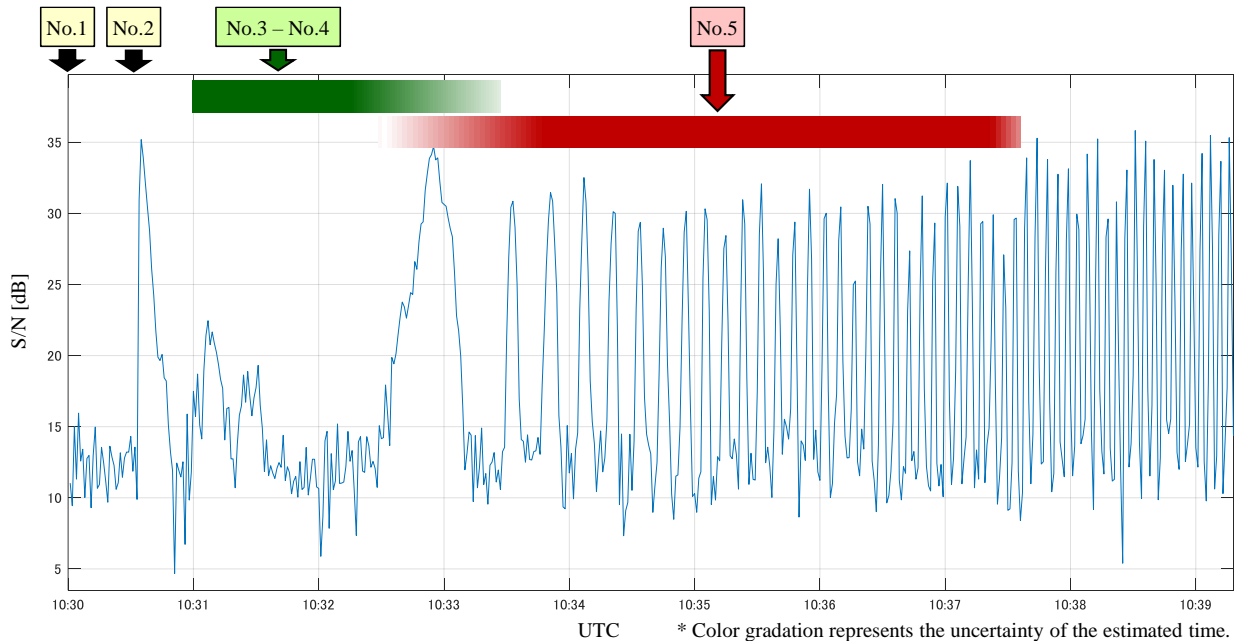


Fig. 13. Reception Signal to Noise Ratio (expanded)

Table 4. Failure scenario

1	OMOTENASHI was released from the dispenser.
2	Power was turned on 30 seconds after separation, and downlink transmission was immediately turned on in an automatic initial sequence.
3	A rate dumping of the initial sequence was initiated because rotation of the spacecraft (due to separation disturbance) deviated from the allowable angular momentum of the reaction wheel. MiPS was injected in the direction to decelerate the rotation.
4	The spacecraft’s rotation converged to steady. The rate dumping was terminated.
5	[Accident occurred] After the rate dumping ended, the spacecraft suddenly began to rotate, accelerating to approximately 80 deg/s in about 5 minutes. Since the rate dumping of the initial sequence had already ended, rate dumping did not start again. XACT failed to acquire the sun due to deviation from the allowed angular momentum of the reaction wheel.
6	Since the radio wave received at the ground station was weak, an uplink was initiated.
7	Downlink transmit power was set to high power, and telemetry was confirmed. The operator recognized that the spacecraft was spinning and not generating power.
8	It took about 10 minutes to understand the situation and discuss the solution. A decision was made to restart the rate dumping by command.
9	The downlink transmit power was set to low power to conserve battery power.
10	Turned off downlink transmission to conserve battery power.
11	The OMOTENASHI team judged that the rate dump process would not be completed in time. The process was stopped.
12	Turned on the downlink and set the output to high power for a telemetry check.
13	Turned off downlink transmission to conserve battery power.
14	The team attempted to direct the SAP toward the sun by injecting MiPS in a direction orthogonal to the rotation. However, the increase in current due to MiPS ON caused an OBC reset. After the reset, the transmitter was turned on at Low Power by the initial sequence, but no telemetry was visible. Since then, an OBC reset, initial sequence start, and MiPS ON for rate dump were probably repeated.
15	The battery voltage fell below that required to drive COM. The downlink signal was lost.
16	The battery voltage fell below the threshold of over-discharge protection, which caused the spacecraft power supply to turn off completely.

4.3. Estimated cause of failure

This section discusses why the angular velocity accelerated from 0 deg/s to 80 deg/s in about 5 minutes. The rate dumping logic was sure to pose no problem as rate dumping control had been completed right after rocket separation. Since the angular momentum was more than that controllable by the reaction wheel, it can be inferred that some external force was generated. The current study assumes an anomaly in the gas jet thrusters of MiPS. The separation disturbance from the rocket and a collision with other secondary payloads are unlikely as the rotation gradually accelerated. As already mentioned in Subsection 3.2.4, there is liquid propellant in the plenum of MiPS. After the first rate dumping was completed, a thruster valve presumably did not close properly for some reason, thereby releasing liquid propellant.

The thruster valves always closed properly in the ground tests. Possible reasons for the non-closing of the thruster valves include contamination or deterioration of the valve seal, which is stored with liquid propellant for three years.

5. Future operation plan

5.1. Future prediction of spacecraft status

First, this section discusses the attitude state of OMOTENASHI. The spacecraft was rotating around its Y-axis when losing the signal. Since the Y axis of the spacecraft is the maximum principal axis of inertia and stable in attitude dynamics, the spacecraft is expected to maintain its Y axis rotation. Moreover, the rotation axis is maintained in the inertial coordinate system due to the law of conservation of angular momentum. Next, this section considers the orbit of OMOTENASHI. After losing signal, OMOTENASHI proceeded along a separation orbit from the launch vehicle and then a lunar swing-by, much like Orion and the SLS Interim Cryogenic Propulsion Stage (ICPS) did. Since then, the spacecraft currently revolves around the sun, gradually moving away from Earth. As the spacecraft moves along its solar orbit, the sun's direction changes so that the solar panels will be exposed to sunlight in the future. Should the solar panels generate electricity, there is a possibility that the power supply of OMOTENASHI may be turned on again.

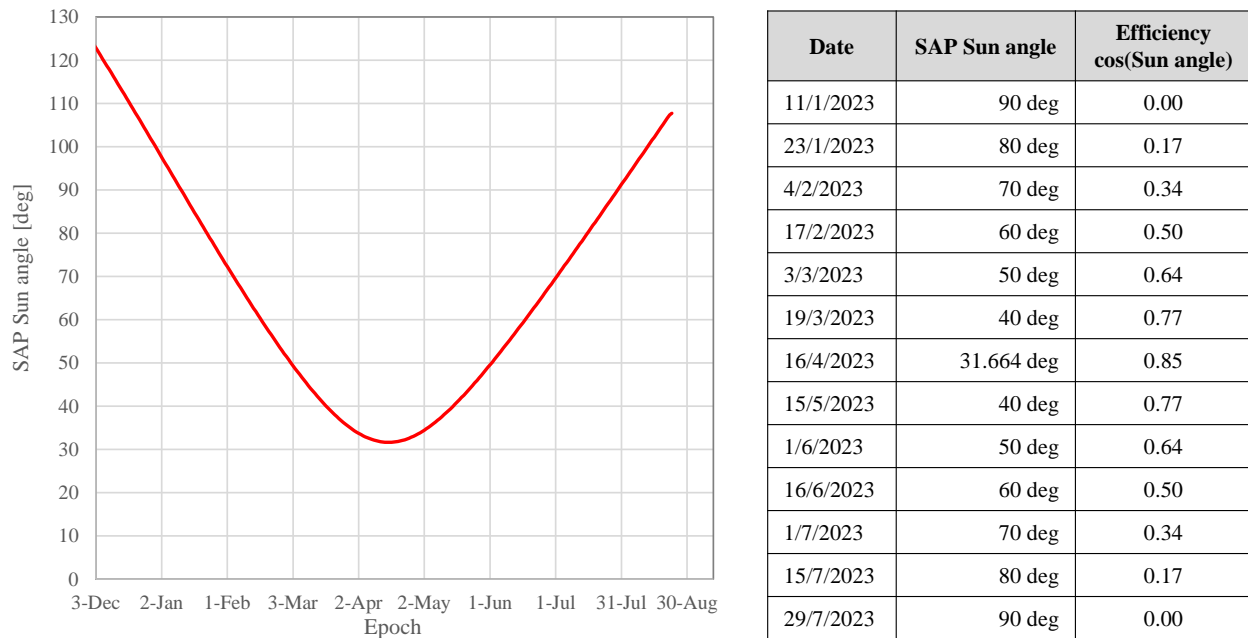


Fig. 14. SAP sun angle

In terms of heat, the solar panel is a heat-absorbing surface whereas other surfaces are reflective. If the solar panels are not oriented toward the sun, the entire spacecraft system becomes very cold. In particular, the battery temperature has been analyzed to reach -70°C outside the guaranteed temperature, causing the electrolyte to freeze. The battery should be considered in cases of both permanent failure and recovery. Even if the battery has failed permanently, the spacecraft system can still be started by using only the power generated by the solar panels.

The state of the spacecraft changes as the sunlight angle to the solar panels increases. It is also affected by whether the battery has failed. The operational simulator investigated the future state of the spacecraft. The amount of power generated by the solar panels was gradually increased in each case. The simulator showed that the spacecraft could be categorized into four conditions as shown in Figure 15. The power generated by the solar panel is still weak in state I, and the OBC is restarted immediately after activation. Therefore, no downlink radio signals are transmitted, and no command and telemetry operations can be performed in state I. Then, when the generated power increases, the OBC is turned on and an initial sequence is started in state II. The downlink signal begins to transmit. As the spacecraft rotates, a rate dumping is automatically executed at 30 seconds of the initial sequence. However, the high startup power consumption of MiPS causes a lack of electricity and an OBC reset. After the reset, the initial sequence of the OBC is restarted. OBC power-on and reset are repeated every 30 seconds. If the battery has failed permanently, the spacecraft maintains state II, even with an optimum sunlight angle of the solar panels. If the battery has recovered, the spacecraft can be in state III or IV. In state III, the OBC will no longer be reset due to the power consumption of MiPS. However, MiPS will not operate properly due to an insufficient power supply. In this case, the software can continue the rate dumping process, but the rotation is maintained because thruster injection is not performed. The system condition cannot be changed from state III to state IV because an OBC reset does not occur in state III. If the condition transitions from state II to state IV, rate dumping is performed due to normal MiPS operation, and solar acquisition control is performed after rate dumping. Finally, the spacecraft rotates at a slow speed around the solar axis for a period of 720 seconds in state IV.

state	spacecraft condition	season			downlink	OBC steady Power-on	rotation	attitude	strategy for search operation
		17 Feb.	3 Mar.	19 Mar.					
I	OBC reset occurred immediately after OBC is turned on.	+	+	-	-	-	7.2 s	the same at loss of signal	Inoperable
II	OBC reset occurred immediately after MiPS is turned on.	-	+	+	+	- OBC reset every 30 seconds	7.2 s	the same at loss of signal	TEST MODE command in order not to turn on MiPS
III	Rate dumping cannot be completed due to lack of power supply to MiPS	-	+	+	+	+	7.2 s	the same at loss of signal	
IV	Rate dumping is completed	-	+	+	+	+	720 s	sun pointing	Rotation acceleration commands up to 120 s period Downlink antenna switch command

+: Possible, -: Not Possible

Fig. 15. Future spacecraft states

5.2. Search operation

It is not possible to explore the spacecraft in state I as listed in Figure 15. Therefore, exploration is expected to begin after March 2023 when the spacecraft will be in states II, III and IV. In fact, exploration operations are planned to start in February 2023, including a margin period. As it is not possible to determine whether the spacecraft is in state II, III or IV, an operation plan that enables exploration regardless of the status must be prepared. In the case of state II, MiPS can be prevented from power-on by the TEST MODE command that which was originally used in ground tests (described in Subsection 2.1.2). When MiPS is not activated, an OBC reset will not occur in state II and radio signals can be obtained stably. In state IV, there is a risk of missing the downlink radio signal if the antenna is facing the direction opposite of the earth due to the slow spin of the 720-second cycle. Therefore, acceleration commands for a sun-directed spin by the reaction wheels and antenna switching command are effective. And in all spacecraft states, a high-power downlink and a suspension of telemetry modulation are both effective for finding radio signals. Based on the above, details of the operation plan are now being considered.

5.3. Similar cases

Recovery operations for SOHO [11] and Hayabusa [12] are cases similar to the current OMOTENASHI status. SOHO was a solar observation satellite jointly developed by NASA and ESA, and was in a halo orbit around the L1 Lagrangian point between the Sun and Earth. In 1998, SOHO suddenly could not control its attitude due to failure of its attitude sensor, and then experienced a loss of communications and battery depletion. By changing the sun’s direction due to orbital motion, it was possible to start generating and restoring electricity. Hayabusa was an asteroid probe developed by JAXA. In 2006, during its return to Earth from the asteroid, a thruster failure due to a fuel leak caused attitude disturbance, resulting in a loss of communications and battery depletion. It was possible to start generating and restoring electricity again because the sun’s direction changed as in the SOHO situation. There are two differences between these precedents and OMOTENASHI. The first point is the accuracy of orbit determination; both SOHO and Hayabusa were able to determine their orbits with high accuracy and did not need to sweep their antennas to search. Conversely, OMOTENASHI had almost no orbit determination measurements. Since the ICPS released a CubeSat while spinning, the separation speed and direction from the dispenser are unclear. Moreover, the large influence of the lunar swing-by caused a huge error in orbit determination. Therefore, it is necessary to search a vast area with a sweeping antenna. The second point is the communication distance. Because both SOHO and Hayabusa never deviated from their original orbits, communication could be established again when power was restored and the antenna pointed to Earth. However, the communication distance of OMOTENASHI was designed based on a distance to the Moon, and the spacecraft is expected to exit the range of communication around summer 2023. An efficient orbital search method is currently being studied to locate the spacecraft while it remains within an effective communication distance.

5.4. Operations after recovery

After restoration, several observations and experiments are being considered, such as continuing radiation measurements outside Earth’s magnetosphere, solid rocket motor ignition experiments, checking the power generated by thin-film solar cells, and checking the performance of the attitude control system.

6. Conclusions

OMOTENASHI's nominal operation plan was developed based on the severe constraints of the 6U CubeSat and its short five-day operation period. In the nominal operation plan, onboard software and operation methods were devised to reduce the risk of human error. Nominal operation could not be conducted due to a malfunction that occurred immediately after separation of the launch vehicle. When the telemetry was received, the spacecraft was rotating at high speed with the solar panels facing the direction opposite to the sun. Finally, the batteries of the spacecraft were depleted and the spacecraft's signal was lost. Leaking liquid propellant from a thruster valve was deemed the cause of the malfunction. Since the spacecraft goes along its sun orbit, the solar panels may be exposed to enough sunlight to reactivate the spacecraft system. Command plans of a search operation are being considered.

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