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Structural Design of De-orbit Mechanism Demonstration CubeSat FREEDOM

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With increasing number of microsatellite launches, the mitigation of space debris is becoming a pressing problem. An easy-handling de-orbit device is required to deal with this issue. However, existing devices face some problems when being installed in a microsatellite. These problems include the volume, direction in which the booms are extended, and battery consumption. Since 2010, Tohoku University and Nakashimada Engineering Works, Ltd., have been developing a De-Orbit Mechanism (DOM) that deploys a thin film to drag microsatellites down to reenter the atmosphere. A DOM deployment procedure was devised to achieve light weight and low power consumption. For an early orbit demonstration of the DOM technology, Tohoku University and Nakashimada Engineering Works, Ltd., are developing a single-unit-size CubeSat named "FREEDOM," which aims to demonstrate a DOM. The DOM for FREEDOM is capable of deploying a thin square film with edge lengths of 1500 mm out of a total satellite mass of less than 1.33 kg. The film size vs. satellite weight ratio of the FREEDOM will be the highest in Japan and is one of the highest in the world. FREEDOM is planned to be delivered to the International Space Station by the Japan Aerospace Exploration Agency and will be deployed into Low Earth Orbit (LEO) in 2016. This paper describes the structural design and verification results of FREEDOM.

Key Words: CubeSat, Microsatellite, Space Debris, De-orbit, International Space Station

Nomenclature

MSy	:	yield stress margin of safety [-]
MSu	:	ultimate stress margin of safety [-]
P	:	calculated stress [MPa]
Py	:	yield stress of the material [MPa]
Pu	:	ultimate stress of the material [MPa]
FSy	:	yield stress factor of safety (= 1.5) [-]
FSu	:	ultimate stress factor of safety (= 2.0) [-]
G	:	acceleration [G]
f	:	frequency [Hz]
PSD	:	Power Spectral Density [G^2/Hz]
Q	:	Q factor [-]
Subscripts		
x	:	X-axis
y	:	Y-axis
z	:	Z-axis
max	:	maximum

1. Introduction

In 2007, the Inter-Agency Space Debris Coordination Committee published Space Debris Mitigation Guidelines[1)] to demonstrate a consensus for sustainable space development. To reduce space debris in the future, satellites in Low Earth Orbit (LEO) should be de-orbited or maneuvered into a short-lifetime orbit. According to the guideline, 25 years is a reasonable and appropriate lifetime. Some mitigation measures using drag sails have been studied by many researchers and developers.

The most common idea is to use extension booms to pull out and deploy stored thin films from a container installed on the spacecraft. Nanosail-D[2)] and Nanosail-D2[3)] of NASA LightSail-1[4)] of The Planetary Society, Deorbitsail[5–7)] from the University of Surrey, and CubeSail[8,9)] from Surrey Space Centre introduced a combination of four triangle thin films and four extension booms to deploy a square-shaped sail out of a 3U CubeSat, whereas the main purpose of the CubeSail was a solar sail mission. AEOLDOS[10)] from the University of Glasgow utilizes a similar mechanism but targets even smaller systems such as 1U CubeSat; thus, the area of the sail is smaller. CanX-7[11)] from the University of Toronto suggested the modular construction of a square sail by combining four units of triangle-sail deployment mechanisms out of the 3U CubeSat. This system consists of four triangle films and eight extension booms.

Although all these devices are useful for CubeSats, they have a common drawback: it is difficult to install them inside the main structures of larger satellites. Thus, the 150-kg-class satellite TechDemoSat-1[12,13)] introduced a drag-sail deployment mechanism that is integrated with the solar panel. This mechanism utilizes four extension booms and four trapezoidal films. Because the size and shape of the mechanism is dependent on the design of the solar panel, this type of mechanism is custom-made for each spacecraft. Some sail deployment mechanisms have, on the contrary, a single square sail instead of separate triangle films.[14–16)] In addition, different types of deployment force generation methods exist; they use booms rolled around the spindle with the sail material. Examples include the centrifugal force demonstrated by IKAROS[15)] and circumferential force.[16)]

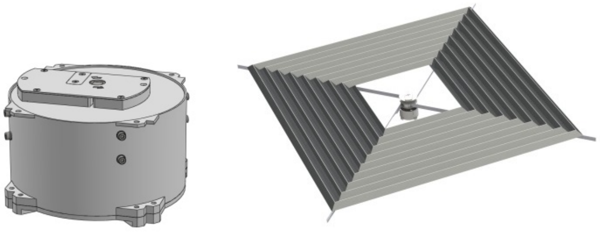

Fig. 1. Appearance of DOM in stored (left) and deployed (right) configurations.

2. Design Objectives for Suggested De-orbit Mechanism

Tohoku University has been developing and operating micro- and nanosatellites for years. These satellites include a 2U-sized CubeSat named RAIKO and the 50-kg-class scientific Earth observation microsatellites RISING-1 and RISING-2.[17–19] Based on these experiences, it has become clear that there should be de-orbit devices for both micro- and nanosatellites for the safe and sustainable utilization of space. In 2010, Tohoku University and Nakashimada Engineering Works, Ltd., initiated the development of a De-Orbit Mechanism (DOM) that deploys a thin film and drags the microsatellite down to reenter the earth's atmosphere. The requirements for the DOM are as follows:

1) It shall be easily installable inside or on the surfaces of spacecraft structures.
2) The extension direction of the booms relative to the spacecraft body shall be fixed (no rotational freedom).
3) A clear and simple mechanical and electrical interface between the spacecraft and the activation of the device shall be as easy as possible (no use of motors, and no assumption/requirements for satellite attitude).

2.1. DOM functionalities

Following the guidelines above, the DOM was first designed as a self-standing, versatile reusable component for micro- and nanosatellites. Unlike the previously mentioned drag-sail-deployment demonstration CubeSats whose mechanisms were incorporated with their satellite structures, the DOM introduces a cylindrical container as its main structure. The container is completely independent from the satellite structure. This container provides for easy handling procedures and the possibility of being installed in many satellites, including microsatellites as well as CubeSats. The appearance of the DOM in its stored and deployed configurations are illustrated in Fig. 1. For applications where the satellite envelope is critical, the DOM's capability of being installed inside the satellite structure plays an important role. To enable this, DOM deployment involves two steps, as illustrated in Fig. 2. First, the internal assembly that holds the extension booms and the sail is ejected from its container by means of a telescopic mechanism. At this moment, the assembly is located outside the satellite structure as well in order to acquire enough clearance for sail deployment. The cross section of the telescopic mechanism is hexagonal to constrain axial rotation.

Second, the sail is pulled out of the assembly simply by means of the strain force of the four booms, which are made of tape springs without electrical motors. This deployment process can be triggered just by supplying 5 V of DC electrical power so that the internal holding wire can be cut by the generated heat. A single sail is used, and the method of folding the sail is inherited from the IKAROS project to achieve the densest installation without requiring separation walls in between. These unique features realized a high ratio of sail size to mass, low electric power consumption, and ease of installation in a satellite. The only requirement is that the top plate of the cylindrical container should provide access to open space.

2.2. Design challenges

The most prominent structural feature of the designed DOM is that the entire mass of the internal assembly is held by an internal hexagonal telescopic cantilever. Another example of a telescopic structure for a drag-sail deployment mechanism can be seen in the European Space Agency project Gossamer Deorbiter,[14] which utilizes a massive portion of the structure for the telescope, and is therefore very large. One of the technical challenges of the slender telescope configuration of the DOM is to ensure mechanical strength and stiffness against mechanical launch conditions. Another technical challenge is to successfully demonstrate the deployment of an IKAROS-type folded sail using a boom-driven deployment force. This paper aims to experimentally evaluate the deployment behavior of the suggested mechanism.

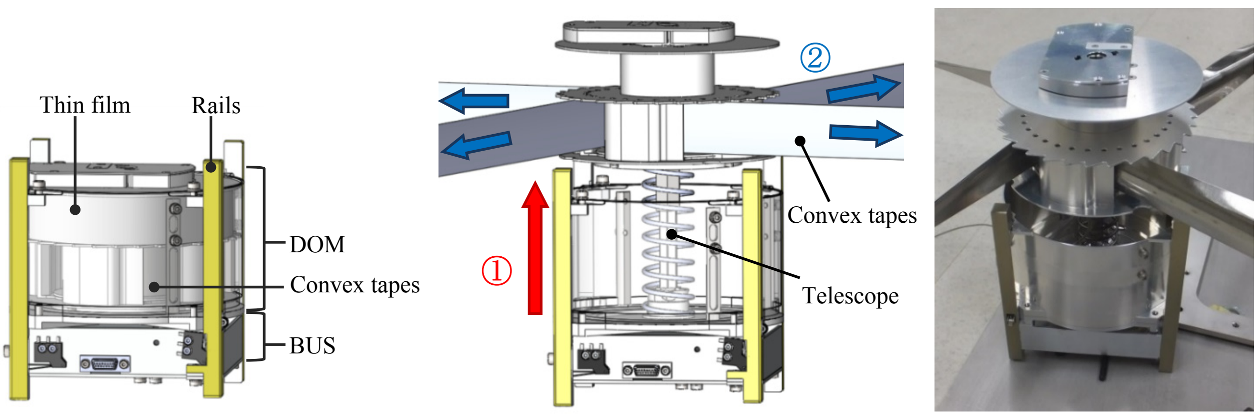

Fig. 2. Mechanism of DOM installed in CubeSat FREEDOM (left: launch configuration, center and right: deployed DOM).

3. System Description of CubeSat FREEDOM

Tohoku University and Nakashimada Engineering Works, Ltd., initiated the development of a 1U CubeSat called FREEDOM for an early orbit demonstration of the DOM. The target DOM is small and lightweight enough to keep the entire mass of the spacecraft within 1.33 kg, which is the standardized limit of a 1U CubeSat. The FREEDOM project aims to monitor orbit transition during the de-orbit, which will measure the functionality of the DOM, and to indicate the usefulness of CubeSats as the orbit technology demonstration infrastructure for relatively large-scale deployable mechanisms. As a result, its film size vs. satellite weight ratio is one of the highest in the world.

3.1. Satellite system

FREEDOM is stored in a release pod called the JEM Small Satellite Orbital Deployer (J-SSOD) and is launched to the International Space Station (ISS). The satellite, which is set on the ISS robot arm, is released from the pod by an astronaut. After the release, the Satellite Central Unit (SCU) starts working and counts time. After a certain time passes, the SCU triggers DOM deployment, and FREEDOM starts to de-orbiting. To purely evaluate the DOM, the satellite is free-flying; hence, no attitude control is installed. The DOM performance is verified by recording the orbital history, which is based on two-line elements data provided by a public organization. The satellite does not require a communication device nor a rechargeable battery system. This not only keeps the system as simple as possible but also represents a real situation where a de-orbit device is activated at the end of the mission lifetime of a spacecraft.

3.2. DOM

The DOM is the mission part of FREEDOM. The DOM has a cylindrical shape and is 72 mm in height and 110 mm in diameter. It has a deployable single square thin film with edge lengths of 1500 mm. A spring and four sets of convex tape deploy the film. These deployable parts are held down by two redundant wires (Dyneema) in the stored configuration. As the deployable part is categorized as a catastrophic hazard, more than one wire is required for the DOM. The wires pass over two heaters with different resistance values (one is nominal, and the other is a redundant resistor). The heaters melt the wires to trigger the deployment. The SCU can monitor the deployment status via a mechanical switch inside the DOM. After the wires are cut, the telescope uses a coil spring to lift the thin film and convex tapes. As the outer case prevents the convex tapes from extending and deploying the thin film, their appearance outside the container by the telescope allows for the deployment to proceed. This contrivance enables the design of lateral satellite structures (e.g., rails) and simple satellite structures without moving parts. The source of the DOM's deploying power is the elastic energy of the spring and convex tapes. No actuators such as motors are needed to deploy the DOM. This leads to a light weight for the satellite by reducing peripheral devices and battery volume. FREEDOM has no solar cells by virtue of its low power consumption. Because of the characteristics described above, the DOM can also be easily mounted on other satellites.

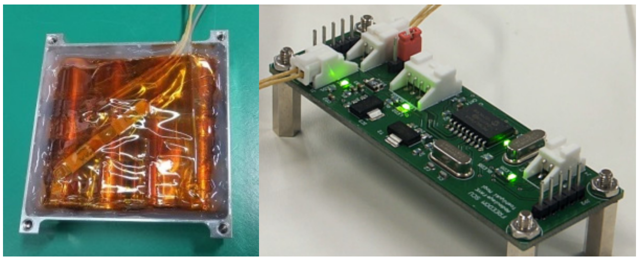

Fig. 3. Battery Unit (left) and Satellite Central Unit (right).

3.3. Battery

Commercial Ni-MH rechargeable batteries are used in the battery assembly (BAT) of FREEDOM. The specifications for the battery cell and BAT are summarized in Table 1, and the engineering model of the BAT is shown in Fig. 3. The BAT is designed considering three characteristics: voltage, weight, and capacity. The wire-cutting heaters require a supply voltage of approximately 5 V. The weight of the BAT is restructured to be less than 160 g. Enough electrical capacity is needed in order to deploy the DOM because there is no charging system in the satellite. Because the BAT is charged only on the ground before launch and will never be charged in orbit, there is no overcharge protection.

3.4. SCU

The SCU (Fig. 3) controls the DOM's deployment timing and switches the nominal and redundant resistors. The SCU is the only electric board in FREEDOM because there is no communication system, attitude control system, solar cells, or actuators. The operating voltage of the processor is set to 3.3 V considering that the BAT voltage is approximately 4.8 V, and will further drop when the DOM resistor is heated.

The SCU starts counting the elapsed time after the satellite is released. There are two transistor switches and two PIC microcontrollers on the SCU, corresponding to the two resistors of the DOM (Fig. 4). When the time count finishes, the SCU turns on the transistor switch to heat the nominal resistor to deploy the DOM. If the DOM cannot deploy for some reason, the SCU turns on the switch of the redundant resistor. The SCU monitors the mechanical switch inside the DOM and turns the switches off when it detects DOM deployment. The timer counting trigger does not need a communication system; therefore, it is applicable to an automatic de-orbit system, which works even if a satellite malfunctions and is no longer able to receive commands.

Table 1. Specifications for battery.

Battery cell	
product name	Eneloop Pro
type	Ni-MH
size	AA (⌀ 14.5 × 50.4 mm)
nominal open circuit voltage	1.2 V
rated capacity	2450 mAh
Battery assembly	
composition	4 series
nominal open circuit voltage	4.8 V
rated capacity	2450 mAh

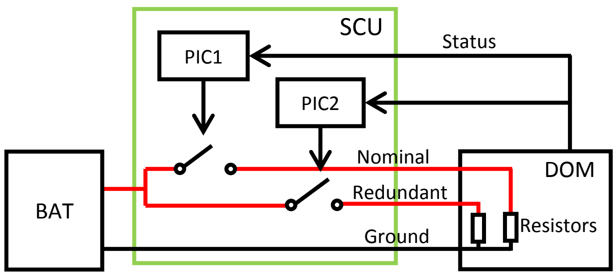

Fig. 4. System diagram of SCU.

4. Structural Design and Analysis

FREEDOM is a single-unit-size CubeSat. The exploded diagram of the structure is shown in Fig. 5. The satellite structure consists of four rails and five panels, which are lightened in order to satisfy the mass requirements, although the structural strength is somewhat reduced. In this project, two characteristics must be ensured before the launch: to withstand the entire FREEDOM structure during launch vibration, and to deploy the DOM in orbit. This section describes the mechanical design and analysis of both the DOM and FREEDOM as integrated into a single CubeSat.

4.1. Structural design

Table 2 shows the weight budget of the satellite, though "Rail+/-Y" contains two rails and a +/-Y face panel, respectively. As the DOM occupies a large percentage (approximately 70%) of the satellite, and the BAT needs enough capacity and voltage, it is inevitable that the weight of the structure must be reduced. To do so, most of the structure is made of 7075 aluminum alloy, and unnecessary parts are omitted. To reduce the number of bolts, some structures are jointly fastened. The satellite does not have an RF communication system, so it is impossible to control the satellite or DOM deployment by up-link commands. For this reason, FREEDOM has three deployment switches, while an ordinary CubeSat released from the ISS has only two switches. Two ordinal switches are installed on the end faces of the rails, and the third switch is installed on the side face of the rail (Fig. 6).

As the satellite needs to be stored in J-SSOD, the dimensional requirement is very strict, and it is desired that the system consists of as few parts as possible. To reduce the number of parts, i.e., to reduce the sources of dimensional errors, each pair of rails in the +/-Y direction are combined into a single-piece structure. Rails are thinned down to reduce weight. As thinned rails warp easily, each rail is attached at three points: two points to the DOM and one point to Panel-Z.

4.2. Structural analysis model

To confirm the structural safety and estimate the trade-off between mass and stiffness, a structural analysis using the Finite Element Method (FEM) was carried out. Femap with NX NASTRAN was the FEM software used for the calculation. Fig. 7 shows the model of FREEDOM used in the analysis. The satellite structure and the DOM are modeled. The number of nodes is 8077, and the number of elements is 5759.

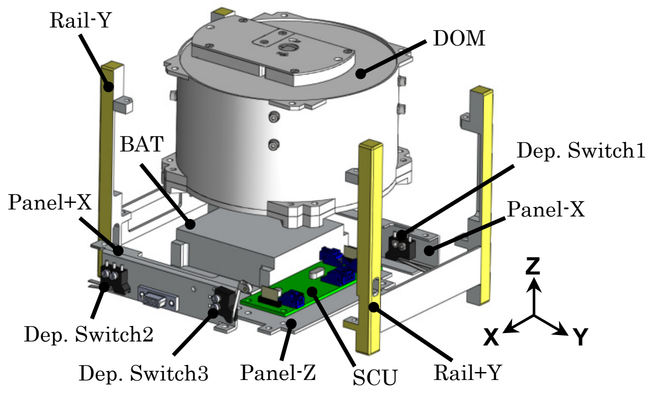

Fig. 5. Exploded diagram of the structure.

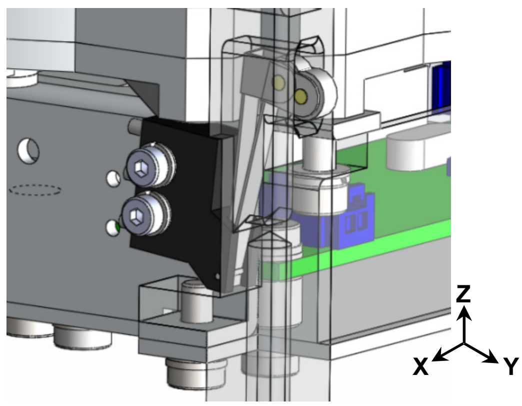

Fig. 6. Deployment switch on rail side face.

4.3. Natural frequency

A natural frequency analysis was carried out to evaluate the stiffness of the satellite. The results are shown in Table 3. Both ends of four rails are set as fixed condition. The minimum natural frequency is required to be over 100 Hz from the launch condition.[20] It is found that the lowest natural frequency is in the X-axis direction, and that all data satisfy the requirements.

Table 2. Weight budget of the satellite.

subsystem	component	parts name	weight [g]
mission 928 g	DOM	DOM	928
power 157 g	BAT	BAT	157
control 50.7 g	SCU	SCU	22.8
		wiring	27.9
structure 159.2 g	STR	Rail+Y	37.9
		Rail-Y	37.0
		Panel+X	9.5
		Panel-X	6.5
		Panel-Z	38.4
		Bolt	29.9
Total (Req.: <1330 g)			1294.9

4.4. Static load

The candidate launch vehicles are following three rockets: H-IIB, Space X Dragon, and Orbital Cygnus.[20] In the analysis, 18.1 G is applied as the largest load among the three vehicles. The yield stress and ultimate stress of the material are shown in Table 4, and the results are shown in Table 5. The shaft located at the center of the DOM and connecting with the deployment part and the structure of the satellite is the most critical part when the vibration direction is along the X- or/and Y-axis. According to the analysis results, the largest von Mises stress and deformation occurs at 1 cm from both ends of the shaft. When the vibration direction is along the Z-axis, the largest von Mises stress and deformation occurs at Panel-Z when the BAT is attached to it. In the result, the margins of safety (*MS*) are calculated as shown below. It is found that the values of *MS* are positive, and that the satellite survives the rocket launch acceleration.

$$MSy = \frac{Py}{P \times FSy} - 1 \tag{1}$$

$$MSu = \frac{Pu}{P \times FSu} - 1 \tag{2}$$

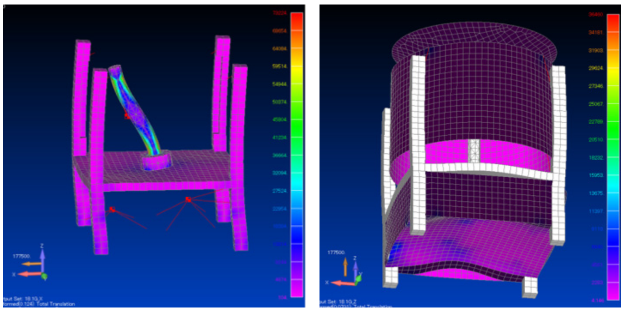

Fig. 7. Results of structural static load analysis.

Table 3. Analysis results of natural frequency.

mode	Axis	Freq. [Hz]
1	X	199.5
2	Y	202.0
3	X	338.4
4	Y	345.0
5	Z	501.6

Table 4. Yield stress and ultimate stress of the material.[21]

part name	material	yield stress[MPa]	ultimate stress[MPa]
shaft	304 stainless steel	179	503
panel-Z	7075 aluminum alloy	475	531

Table 5. Result of static load analysis.

axis	calculated stress [MPa]	*MSy* [-]	*MSu* [-]
X	73.2	1.3	2.6
Y	74.0	1.3	2.5
Z	36.5	7.4	6.5

4.5. Random vibration

Random vibration analysis is also performed to check that the structure withstands random vibrations or the acoustic environment before a random vibration test is conducted. The levels of vibration are listed in Table 6, which shows the envelope of the vibration level for the three rockets.[20] The accelerations applied to the random vibration analysis are calculated by the Miles' equation [Eq. (3)], which is available for a nearly flat input level. This equation is a method to calculate the approximation of a statistical peak load of a random vibration input from the base as a single degree of freedom system. This situation is similar to the random vibration test and predicted to be severer than launch condition. Hence, we applied the peak load and calculated it in the same way as the static load analysis. In Eq. (3), Q was calculated using Eq. (4).[22] The value is 10 for the X- and Y-axes, and 17.5 for the Z-axis. Table 7 shows the results of the analysis. As all results are positive, the structure survives any of the three rockets' launch vibrations and satisfies the requirement of the maximum mass limit.

$$G_{\max} = 3 \times \sqrt{\frac{\pi}{2} \times f \times PSD \times Q} \tag{3}$$

$$Q = \frac{10 + 0.05 f}{2} \tag{4}$$

Table 6. Levels of random vibration.[20]

Frequency [Hz]	PSD [G^2/Hz]
20	0.015
25.6	0.027
30	0.08
80	0.08
113	0.05
400	0.05
1081	0.0045
2000	0.002
RMS	5.9 G_{rms}

Table 7. Analysis result of random vibration.

axis	max random acceleration [G]	calculated max stress [MPa]	*MSy* [-]	*MSu* [-]
X	37.55	113.2	0.05	0.63
Y	37.79	115.2	0.04	0.63
Z	60.02	67.9	3.6	2.8

5. Random Vibration Test

In order to verify the stiffness of the structure and its components, a random vibration test of the satellite was carried out at Yamagata Research Institute of Technology. An Engineering Model (EM) of FREEDOM, which reflects the optimum trade-offs shown by the results of the analysis, is manufactured for this evaluation test (Fig. 8).

The vibration test configuration is shown in Fig. 9. The model is installed in a jig, which is an equivalent configuration to the release pod. The input level of the vibration is the same as the level of the random vibration analysis. The response to the vibration input is measured by monoaxial accelerometers.

From the results of the measurements (Fig. 10), the natural frequencies for each axis are higher than 200 Hz. An inspection was conducted after the vibration test, and it was found that no parts were damaged during the test. During and after the vibration, the DOM did not deploy and remained in the same condition as before the vibration. This suggests that the DOM and the demonstration satellite are of the proper design to survive launch conditions.

6. DOM Deployment Test

DOM deployment tests were carried out before and after the vibration test to confirm the condition of the DOM and peripheral circuit. Before the vibration test, the DOM was deployed as expected. After the first vibration test, however, DOM deployment stopped in midstream because of friction between the telescope elements, which are made of austenitic stainless steel. After investigation, solid lubricants (molybdenum disulfide spray) were applied between the sliding parts of the telescope. This solved the problem (Fig. 11), and the DOM was successfully deployed after a second trial of the vibration test (Fig. 12). It was found that launch vibration can cause sliding parts to stick, and that applying solid lubricants is an effective measure for a successful deployment. A functional test in a vacuum chamber was also carried out on the DOM (without convex tapes), and the telescope with lubricants worked accurately in a vacuum. The satellite temperature range in orbit was expected to be determined from the orbit elements and the satellite's thermal design. The vacuum deployment tests were performed at high and low temperatures: +60 ºC and -20 ºC, respectively.

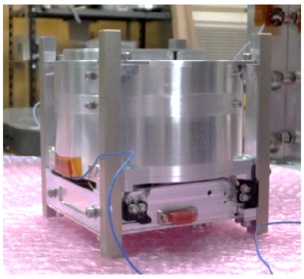

Fig. 8. Engineering model of FREEDOM.

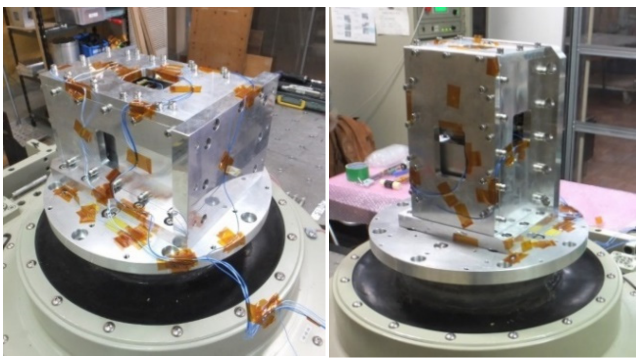

Fig. 9. Vibration test configuration (left: X-axis, right: Z-axis).

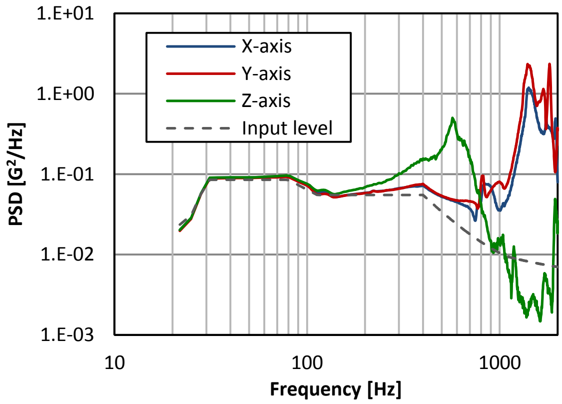

Fig. 10. Random vibration test results.

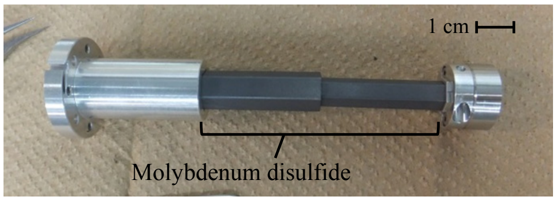

Fig. 11. Telescope mechanism applied solid lubricant.

A sail deployment test at high and low temperatures was carried out using a walk-in temperature and humidity chamber at the Kitakami Industrial Administration Improvement Division. The size of the chamber was 4 m wide, 3 m deep, and 1.9 m high. Sequence photos of the DOM deployment tests are shown in Fig. 12. In all three temperature environments, the DOM was deployed successfully in the same way, and it was confirmed that the device would successfully deploy in orbit.

Figure 12 also illustrates that the film deployment behavior out of the container has two phases. During the first phase, the sail is pulled out of the container but does not spread until approximately 0.33 s has elapsed. In the second phase, the folded sail begins to deploy as the boom extension length increases. Finally, the deployment action is completed within approximately 0.5 s. However, unlike IKAROS, the DOM does not have mechanical equipment to intentionally separate the deployment process into two phases; thus, the behaviors of the sails coincidentally resemble each other.[15] This fact may be a result of using the same folding pattern for the sails. A detailed investigation of this behavior will be the topic of another publication. However, the DOM and IKAROS are different in the ways they generate extension forces. Because the DOM will be used in an end-of-life situation, one cannot assume that the spacecraft can conduct spinning-up attitude control behavior for the sail deployment or for maintaining the shape of the sail during the de-orbiting period. It was revealed that the DOM can deploy its sail without using centrifugal force, and can maintain the shape of the deployed sail via the mechanical boom structure. This fulfills its requirements perfectly.

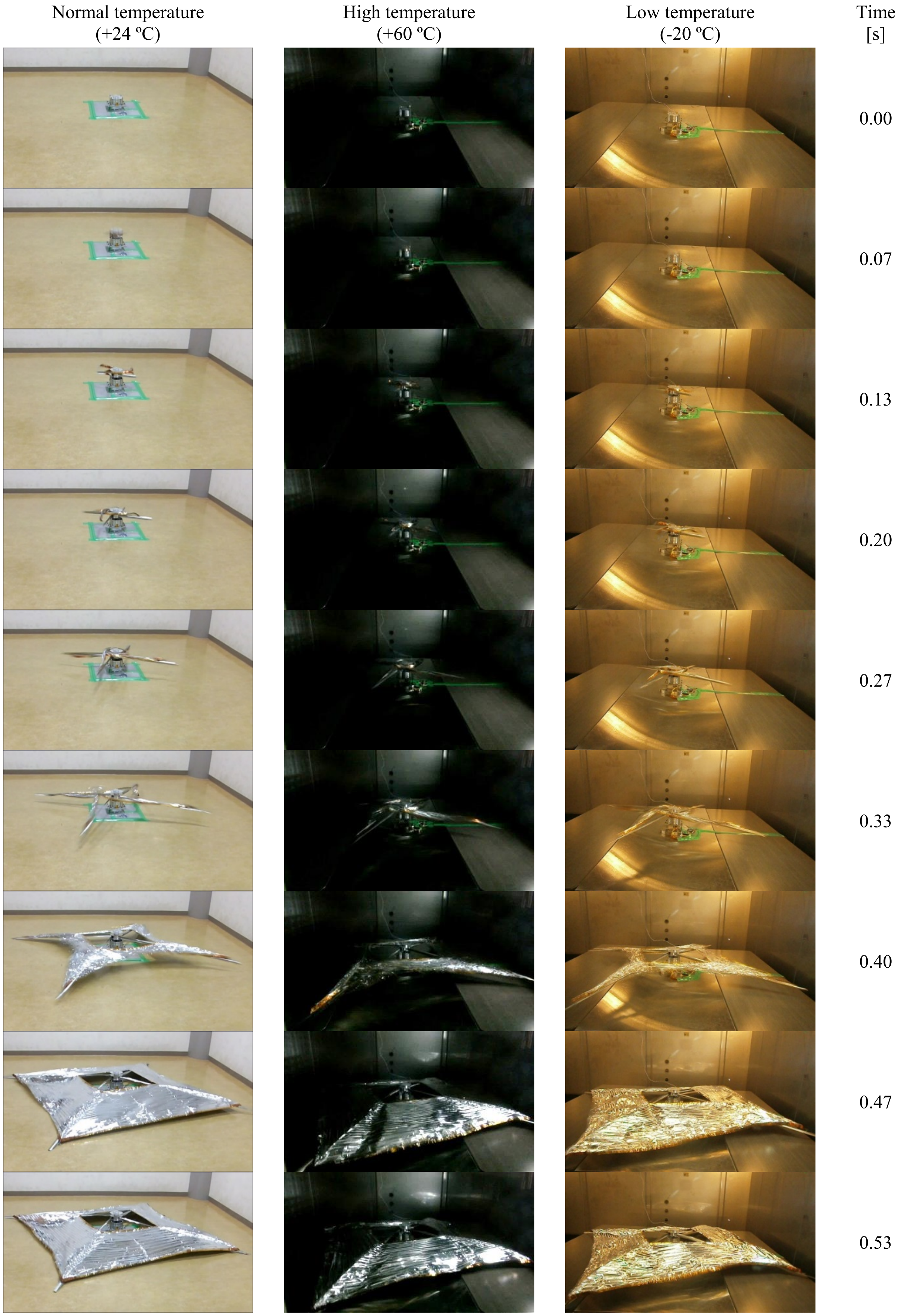

Fig. 12. DOM deployment sequence: after vibration test in room environment (left), high-temperature deployment test (middle), and low-temperature test (right). The light was turned off during the high-temperature test owing to operational limitations of the facility.

7. Conclusion

This paper summarized the structural design and ground verification results for a de-orbit Mechanism (DOM) on the CubeSat FREEDOM. Owing to features of DOM such as the an internal telescopic structure and motorless design, a high ratio of film size vs. satellite weight could be realized. The DOM is an easy-to-use de-orbit device for micro- and nanosatellites. The DOM has advantages in size, mass, power consumption, and installation flexibility. It is found that stress concentration in the presence of launch vibration could cause the deployment failure of the telescopic mechanism. In this research, it is illustrated that enough stiffness and strength could be ensured through an analysis and vibration test. In addition, it is shown that solid lubricant is effective for preventing the sticking of the telescopic structure. Furthermore, deployment tests in high- and low-temperature environments in a vacuum were successfully conducted, and the tolerance of the DOM against the space environment could be verified. Consequently, the mechanical design of the DOM and FREEDOM could be finalized, and development of the flight model for FREEDOM is now being completed. The functionality and performance of the DOM on FREEDOM will be orbit demonstrated in 2016.

Acknowledgments

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