3 Development of the Satellite System

3-1 Development of Optical Inter-orbit Communications Engineering Test Satellite (OICETS)

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Optical Inter-orbit Communications Engineering Test Satellite (OICET) “Kirari” is a satellite developed to demonstrate innovative technologies of laser-based optical inter-orbit communications between OICETS and European Space Agency’s (ESA) Advanced Relay and Technology Mission Satellite (ARTEMIS) and planned to be launched by Japanese J-I launch vehicle from Tanegashima Space Center.

After its Proto Flight Test has been completed, the plan had to be changed. Finally OICETS was launched by Dnepr launch vehicle from the Baikonur Cosmodrome in the Republic of Kazakhstan in August 2005, and succeeded in the world first bi-directional optical communication experiments between two satellites and between low earth orbiting satellite and a ground station.

This paper describes the overview of OICETS and features of its development.

Keywords
Inter-satellite laser communications, Optical Inter-orbit Communications Engineering Test Satellite (OICETS)

1 Introduction

Optical communications are expected to promote, in principle, the downsizing and weight saving of communication equipments, and improve their communication speed. Optical communications have great directivity with less interference than radio waves, and hence have an advantage in transmitting highly-confidential information. It is therefore expected that optical communications can be used for making a compact, light-weight communication equipment mounted in a small or medium-size earth-orbiting satellite or a deep-space exploration spacecraft.

Japan Aerospace Exploration Agency (JAXA) (formerly, National Space Development Agency of Japan) began to survey and examine optical inter-orbit communication equipments in 1985 and made research model prototypes of the equipments between 1991 and 1993. Based on the results, JAXA investigated if it was possible to conduct an in-orbit experiment using ARTEMIS. It then began to develop Kirari to be launched, using the J-I launch vehicle, to a low earth orbit at an altitude of 610 km, with an orbital inclination of 35°.

The project was suspended after the protoflight test (PFT) of Kirari, although periodic tests, tests for life-limited items, and additional tests for installed software were still conducted to maintain or improve reliability.

When ARTEMIS succeeded in reaching geo-stationary orbit using its propulsion system, we brought a Laser Utilizing Communications Equipment (LUCE) engineering model (EM) to an ESA optical ground station on Tenerife Island off the coast of Spain in September 2003, and performed experiments to check the acquisition and tracking of ARTEMIS in its orbit, as well as communication functions.

The project was restarted in 2004 with the launch vehicle switched to a Dnepr rocket and the orbit to a sun-synchronous orbit. Although the situation surrounding the satellite had changed greatly, we did not have to change the basic design of Kirari because it had been designed for a broad range of missions with an aim to make it a standard 500 kg-class bus for J-I launch vehicles. The satellite was then launched successfully in August 2005 and used for various in-orbit experiments until it was switched off in September 2009.

2 Overview of Kirari development

Kirari is a satellite with a launch weight of about 570 kg. The dimensions of the satellite body are 1.1 m wide × 0.78 m long × 1.5 m high (1.7 m including the rocket connection ring). This rectangular parallelepiped shape can fit in the ϕ1.4 fairing of the three-stage J-I launch vehicle and also in the fairing of the Dnepr rocket. The satellite deploys two solar array paddles in orbit to produce necessary power. When deployed, the distance between both ends of the paddles is 9.36 m. Figure 1 shows the appearance of the satellite finalized at the launch site.

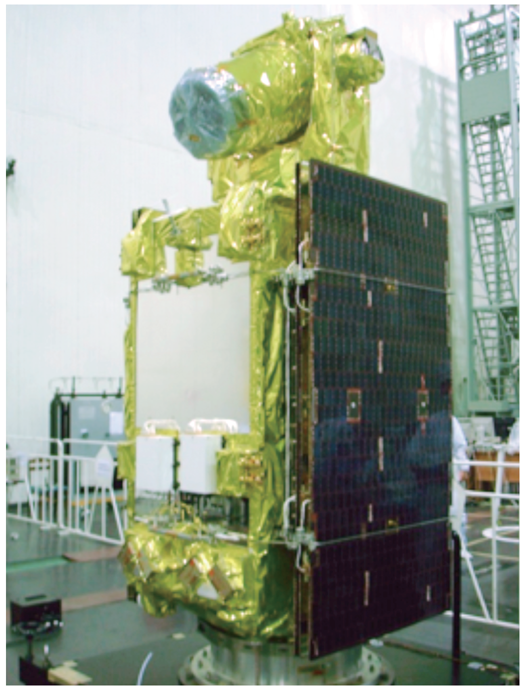

Fig.1 *Final appearance of Kirari before launch*

Kirari usually uses an inertial reference unit (IRU), conical scanning earth sensor (CES), and a fine sun sensor (FSS) for attitude detection, and four reaction wheels (RW) for three-axis stabilized attitude control to keep the bottom of the satellite (rocket connection side) oriented to the earth. The optical unit of the Laser Utilizing Communications Equipment (LUCE-O) and S-band inter-orbit communications antennas are mounted on the upper panel of the satellite for clear communications with ARTEMIS and Japan’s Data Relay Test Satellite (DRTS).

In the communication experiments with the optical ground station, an “inertia lock” mode was used to fix the satellite’s attitude in the inertial reference frame according to IRU measurements. When going half way around the earth in “inertia lock” mode, the satellite points the face where the LUCE-O was mounted to the earth, which allowed for the communication experiments with the optical ground station.

Figure 2 shows a functional block diagram of Kirari, and Table 1 presents the major specifications. The mission payload consists of the Laser Utilizing Communications Equipment (LUCE) and Micro Vibration measurement Equipment (MVE). LUCE has two main units: One is the optical unit (LUCE-O) which acquires, tracks and points a target satellite and sends/receives optical signals, and the other is the electronic circuit unit (LUCE-E) which sends/receives electric signals to/from satellite buses and controls the optical unit. The electronic unit is installed inside the satellite body. LUCE-O consists of a dual-axis gimbal, Cassegrain telescope optical antenna with a 26 cm primary mirror, and inner optical unit in which sensors, laser diodes, and relay optical systems are installed. MVE is used to

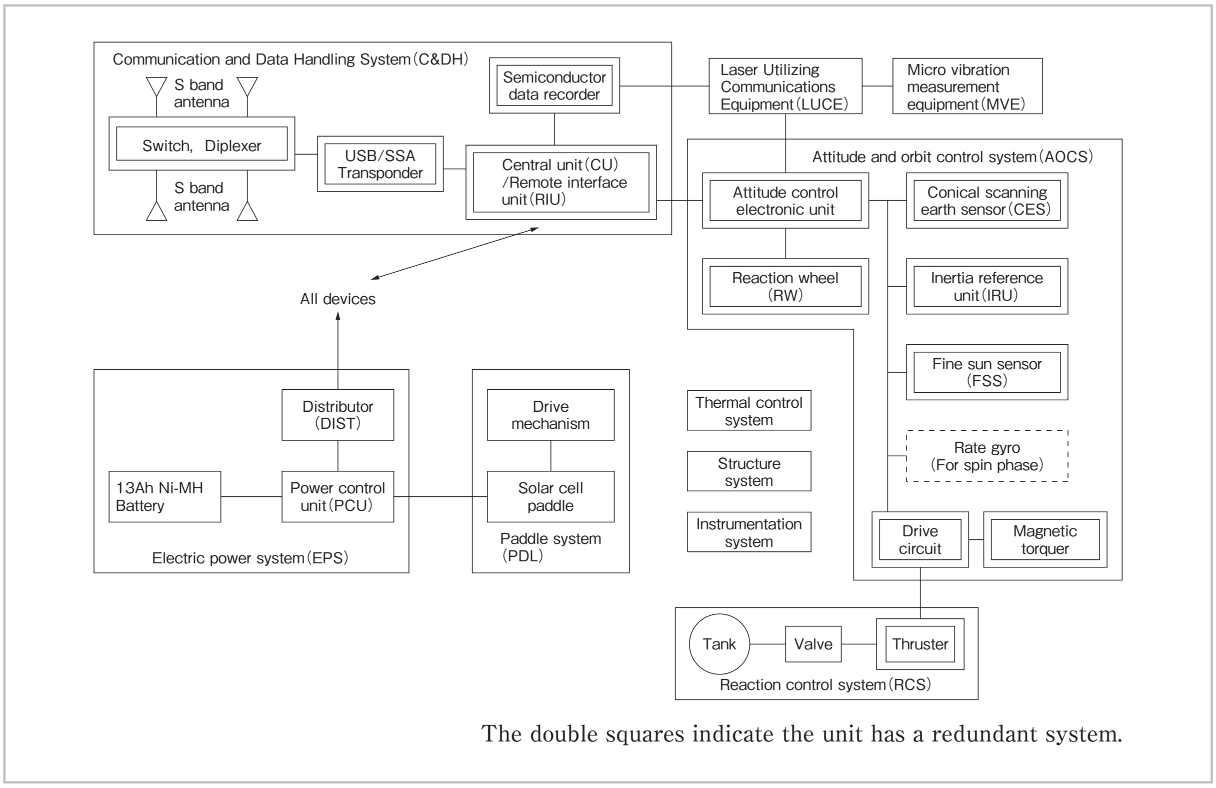

Fig.2 *Functional block diagram of Kirari*

measure micro vibrations produced by the drive system of a reaction wheel, etc. in Kirari, and to evaluate any influence from vibrations on the acquisition and tracking performance of LUCE.

The S band transponder of the communication system has the following modes: USB (Unified S Band) mode which is used mainly for house-keeping telemetry and command transmission from/to a ground station, SSA (S band Single Access) mode which is used for communication with a data relay satellite, and HSB (High-speed S Band) mode which is used mostly for mission data downlinks to a ground station. The attitude control system not only has an earth sensor, a fine sun sensor, and an inertia reference unit, but also a rate gyro, which is used when the solid rocket J-I injects a spinning satellite into orbit. For the actuator, four reaction wheels and three magnetic torquers are used. The reaction control system consists of four sets of thrusters and two tanks. These four sets of thrusters are minimum sets to control the attitude and orbit and so lead to weight and cost saving. On the other hand, since the thrusters are all had to be located only for increasing the speed of the satellite, speed reduction control has to be made by using the “inertia lock” mode to reverse the attitude and orient the thrusters in the desired direction. The tank can contain up to 45 kg of fuel to compensate for rocket injection error. The thermal control system is mostly passive system but provides heater control for the battery, reaction control system, and other equipment whose allowable temperature ranges are critical.

In order to make these units and their redundant systems compact, and to maintain their accessibility, the main body structure is designed in such a way that the two side panels (+X and −X panels), which have no solar cell paddles, can be opened.

3 Development of Kirari

To reduce the development costs of Kirari, a two-step development was employed: the

Table 1 *Major specifications of Kirari*

Item	Specifications
Weight	About 570 kg
Shape	Box shape with two solar cell paddles
Dimensions	Satellite body: 1.1 m × 0.78 m × (height)1.5 m Height with LUCE at top: 2.93 m Width when paddles are opened: 9.36 m
Solar cell paddle system	Power generation: 1220 W or higher (EOL, β angle = 0°) Highly-efficient NRS/BSF silicon cell
Power system	Astable bus-type distributed power system 13Ah, Two Ni-MH cells in parallel
Attitude control system	Strapdown attitude determination system 4-skew zero momentum attitude control system
Communication data processing system	USB/SSA shared transponder Semiconductor data recorder
Reaction control system	Hydrazine monopropellant blow down type 1N thruster × 4 × 2 (full back-up system)
Instrumentation system	Laser reflector (CCR)
Mission unit	Laser Utilizing Communications Equipment (LUCE) Micro Vibration measurement Equipment (MVE)
Mission term	1 year or longer
Orbit	Altitude: 610 km Orbital inclination: 97.8° (sun-synchronous orbit)
Launch vehicle	Dnepr rocket
Launch site	Baikonur Space Center
Launch date	August 24, 2005

development of a ground test model and an orbiting model. We therefore used ready-made components, if available, for the satellite bus and decided not to make a BBM- or EM-like model except that some components needed to be newly developed, such as the semiconductor recorder and 13Ah Ni-MH battery.

We had significant difficulty in developing the mission unit LUCE and took the following actions in relation to major problems faced:

1) LUCE was developed with original Japanese technology, according to the documentation for the interface with the Optical Payload for Intersatellite Link Experiments (OPALE) mounted on ARTEMIS. Due to the development schedules of the LUCE and OPALE, it was planned not to check the interface using actual units on the ground. We therefore checked the interface in the following manner.

-The effectiveness of the communication sequence, which was important to establish an optical link, was checked by performing simulations based on a mathematical model created by Japan and ESA.

-The conformity of the optical properties required for the interface, such as wave length and polarization, was confirmed through a LUCE development test using the same optical property test equipment as that used for the development test of OPALE. The test equipment was able to measure wave length, far field pattern, polarization, and other optical properties in both the air and in a vacuum.

-An acquisition and tracking test for ARTEMIS in geostationary orbit was performed using a LUCE engineering model mounted at the ESA optical ground station. Although it could not simulate the effect of the attitude disturbances and thermal distortion that could occur in the space environment, the ARTEMIS acquisition and tracking function, optical communication function, and point ahead function were able to be checked through this test.

2) In order to develop a high-precision optical system to be used in the space environment, we needed to develop a design that avoided the influence of thermal distortion as much as possible. In particular, for the optical antenna, the distance between primary and secondary mirrors needed to be maintained to an accuracy of several micrometers to meet the requirement for the wave front error. Also for the antenna, a passive-type thermal control was employed without using a heater or other active-type devices. The primary and second-

ary mirrors and telescope tube were all made of special glass materials that have an extremely small coefficient of thermal expansion at normal temperature. Also, to prevent strain due to the connecting elements of the equipment, the elements were cut, polished, and bonded for connection. The surface of the primary mirror was polished to an accuracy of $\lambda/20$ (rms) or smaller to meet the requirement for the wave front error.

Since the optical antenna was made of fragile materials, the strength against the load (quasi-static acceleration) that would be applied to the antenna during tests on the ground and the launch needed to be assured. From the viewpoint of fracture mechanics, a micro crack that is created in the manufacturing process becomes larger due to the load given off during tests on the ground and the launch and finally breaks the optical antenna. Since it is difficult to visually check the presence/absence of such micro cracks, we decided to confirm the strength by applying a certain load to the optical antenna and making sure that it would not be destroyed. For this proof test we set the optical antenna in a vibration test machine and applied a sine burst to the antenna. The sine burst was a short-lasting sine wave in two or three periods. Figure 3 shows a photograph of the test.

3) Kirari is a relatively small satellite but has the characteristic of having two control systems for its missions. One is an attitude control system and the other is a control system for LUCE. The LUCE control system, using its own sensor output, orients LUCE-O mounted on the satellite to ARTEMIS while the attitude control system keeps the satellite oriented to the earth.

The LUCE control system consists of a coarse tracking and pointing system (CP system), which has a CCD as a light-receiving sensor and a direct drive motor to drive a biaxial gimbal, a fine tracking and pointing system (FP system), which uses a four-quadrant photo detector as a light-receiving sensor, a multi-layer Piezo device to drive the mirror, and a point ahead system (PA system), which has the same configuration as the FP system.

For the achievement of the mission of the optical inter-orbit communications engineering test, it was critical for these control systems to work collaboratively, deliver high performance under disturbance on each control system, and prevent mutual interference or interference with the satellite dynamics. Figure 4 illustrates the mutual relationship.

In the design of the satellite dynamics, we tried to prevent eigenfrequencies from the solar cell paddles or other flexible appendages. However, when we operated LUCE in a proto-flight test, we found a vibration in the satellite body. It was then clarified by simulations that a self-excited vibration could occur with certain satellite configurations and certain con-

Fig.3 *Proof test for optical antenna*

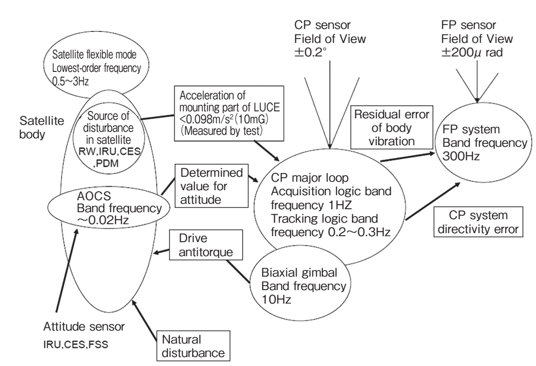

Fig.4 *Diagram of influences of Kirari control systems*

straints (satellite fixing method). We therefore used a filter for the LUCE control system. But the phenomenon occurred under very limited conditions and it was difficult to predict the phenomenon at the design stage. The satellite did not display this phenomenon in orbit and worked correctly.

The drive units of Kirari in Fig. 4, i.e. reaction wheel (RW), paddle drive machine (PDM), conical scanning earth sensor (CES), inertia reference unit (IRU), and LUCE, can generate internal disturbances. Since it is difficult to analytically evaluate the influence of disturbances on the stability of LUCE's directivity, we performed an acquisition and tracking test in the presence of micro vibrations using the equivalent hardware as the actual units.

In this test, as shown in Fig. 5, a low-eigenfrequency suspender is used to suspend the satellite. A vibration stopper (foam: expanded polyethylene) was put beneath the suspender and satellite. We tried to suppress as much as possible the vibration sources that could affect measurement. For example, we stopped the air conditioner and conducted the test at nighttime to avoid vibrations due to vehicles around the building. By doing this we succeeded in suppressing the background disturbance acceleration to 10^{-2} m/s^2 (about 1 mG$_{o\text{-}p}$) or lower and could make sufficient measurements to determine the influence of the micro vibrations.

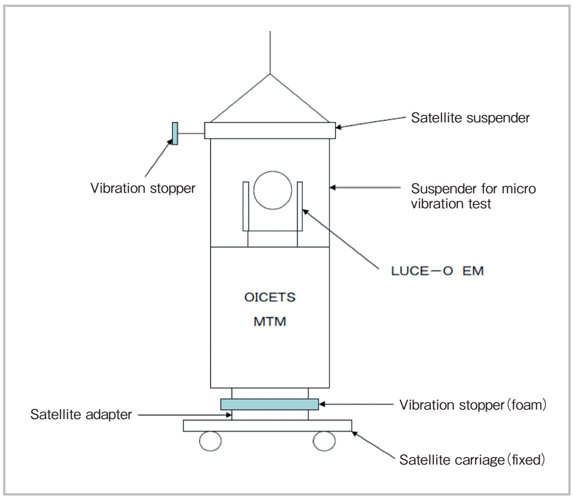

Fig.5 Configuration of acquisition and tracking test in the presence of micro vibrations

For the evaluation of the directivity of LUCE, the light transmitted from LUCE was received at an opposed terminal simulator (TS) for optical communication to measure the error angle of the optical axis of the transmitted light. The micro vibrations of each component were measured with a highly-sensitive acceleration sensor. As a result we found that the RW was the largest disturbance source and that LUCE met the requirements for acquisition and tracking performance, even in the presence of the largest micro vibrations that one could expect.

In Figure 6, the transfer functions obtained in ground tests and those measured by MVE on orbit were compared under the assumption that the disturbance generated by the four RWs was equal to the acceleration of the mounting part of the IRU measured in the ground test, and that the disturbance in the ground test and that in the on-orbit experiment were of the same level.

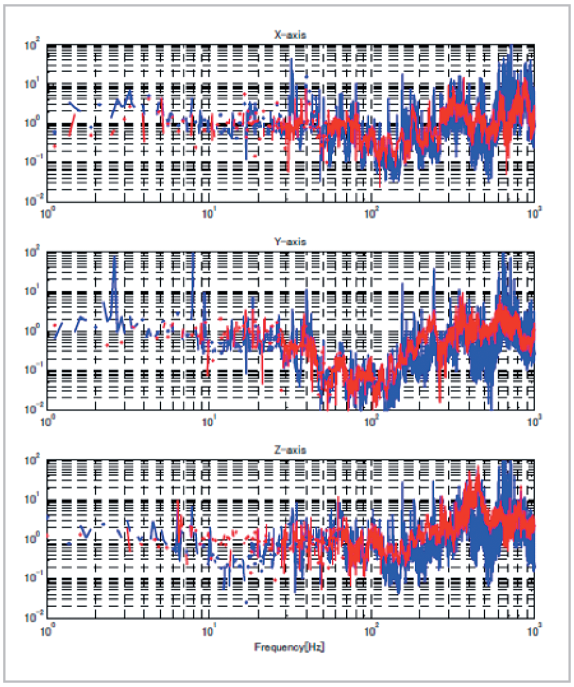

Fig.6 Comparison of communication characteristics obtained in the ground test and in the experiment in orbit

(The red data was obtained from the ground test and the blue from the in-orbit experiment.)

We consequently found that, while the transfer functions obtained in the ground test and those on orbit showed a similar tendency at low frequencies, they were different at high frequencies. The influence of internal disturbances is a recent problem for earth observation satellites and astronomy satellites. For the evaluation of this influence, it is becoming more important to collect measurement data of ground tests and in-orbit experiments. The results that we obtained for Kirari will contribute to this.

4 Launch of Kirari

To have more launch opportunities to restart the project, we allowed a sun-synchronous orbit (of orbital inclination 97.8°) and considered using foreign rockets. As a result of surveys, we decided to use a Dnepr rocket, made in Ukraine, to launch Kirari in July 2004. Table 2 presents the major influences of the change from J-I to Dnepr. For Kirari and the ground system (i.e. tracking control system and experiment planning system), most work was focused on modifying the software for the change of orbit. On the other hand, a change of interface was necessary due to the change of rocket and launch site, mostly because the launch method was changed. Cold launch is usually employed for Dnepr rockets, where the satellite is powered off before the launch and on after separating the satellite from the rocket. For Japanese rockets, hot launch is used, where the satellite power is kept on during the launch. Thus a power supply line (umbilical line) to the satellite, and a communication line with the satellite (RF link), as well as a fairing air conditioner are necessary for a hot launch. We therefore renovated the launch facilities. Also, since the separation technique was changed from the clamp band method used for J-I to the separation bolt method, an interface ring was inserted between the satellite and the satellite separation part, and a mechanism for pulling the umbilical connector at the timing of separating the satellite from the rocket was added. However the satellite structure was not

Table 2 *Influence of change of launch method*

Item	J-I	Dnepr	Action
Orbital inclination	35°	97.8°	"Kirari" -Modification of the attitude control software -Modification of the ground system software (tracking control system and experiment planning system)
Access to satellite after it is loaded in rocket	Yes	No	"Kirari" -Reflected to site adjustment procedure
Launch mode	Hot launch	Cold launch (standard)	"Dnepr" Action taken for hot launch -Umbilical line added -RF link added -Faring air conditioner added
	Spinning injection	3-axis injection	"Kirari" -Modification of the attitude control software
Separation method	Clamp band	Separation bolt	"Dnepr" -Production of satellite interface ring -New production of PAF -Production of mechanism for pulling umbilical connector
Launch site	Tanegashima Space Center	Baikonur Space Center	-New installation of 100 V power supply -New installation of RF link -New installation of LAN -New facilities for high-pressure gas supply -Request of filling propellant to local company

changed since these changes could be absorbed in the design change of the rocket.

A year after this, the satellite and ground facilities (the tracking system and experiment planning system) were tested. We then delivered Kirari to Baikonur Space Center in the beginning of June 2005 and started a launch campaign at the launch site which lasted for two and a half months.

The launch campaign was planned according to a local survey. For example, we requested a local company which knew the local facilities well to fill the propellant (hydrazine), which was imported from Japan to take advantage of the development test results of the reaction control system.

It was also planned to launch simultaneously a small satellite, “Reimei” from the Institute of Space and Astronautical Science, as a sub-satellite. Therefore during the launch campaign, an average of about 30 Japanese people stayed at the site. In particular there were about 50 Japanese people in the launch readiness review. The weather in the daytime in June was very hot and dry with the temperature exceeding 40 Celsius degrees. People also found it difficult to adjust themselves to the meals and other living environments. The differences in language and customs were also a problem when they working collaboratively with local supporting companies at the Baikonur Space Center. Although we had these many difficulties, Kirari was launched on August 24, 2005 with the cooperation of local and Japanese staff (Fig. 7).

Fig.7 Launch of Kirari

5 Conclusions

It took 13 years from the start of the development of Kirari to the on-orbit experiment. We had various problems in the development of LUCE including the change of rocket and launch site. However since we succeeded in the on-orbit experiments, we were able to contribute to research for the demonstration and actual operation of the optical inter-orbit communications.

The author would like to express his sincere thanks to NEC, NEC Toshiba Space System, IHI Aerospace, IHI, ISC Kosmotras, Yuzhynoe SDO, Sumitomo Corporation, and Kokusai AeroMarine for their contributions to the development and launch of Kirari.

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(Accepted March 14, 2012)

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Spacecraft Structure, Structure Dynamics, Pointing Error Caused by Micro-Vibration