FLIGHT RESULTS OF REDUNDANT MEMS IMU (MARIN) ON SATELLITE AND LAUNCH VEHICLE

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ABSTRACT

As the accuracy of MEMS gyroscopes and accelerometers has improved in recent years, there is an increasing possibility of realizing a low-cost Inertial Measurement Unit (IMU) using MEMS gyroscopes and accelerometers for navigation of spacecraft and launch vehicles. Since IMU for spacecraft and launch vehicles needs to be resistant to severe environments, such as high vibration and shock, a wide temperature range, and severe cosmic radiation, the Redundant MEMS IMU (MARIN) was designed to operate and maintain highly accurate inertial measurements in such severe environments.

This paper describes the issues involved in MEMS IMU for spacecraft and launch vehicles, the design and development of MARIN, the results of its in-orbit experience on the RAISE-2 satellite, and the results of its flight experience results during the launch phase on Epsilon rocket F6 and H3 rocket TF1.

1 INTRODUCTION

The Inertial Measurement Unit (IMU) equipped with gyroscopes and accelerometers is one of the most important components for the guidance, navigation, and control system of spacecraft including launch vehicles and satellites. The inertial sensors that provide information about the spacecraft's angular rate and acceleration require high accuracy, high reliability, and environmental resistance. Since spacecraft are subjected to a unique, severe environment with high vibration and shock, wide temperature ranges, and severe cosmic radiation, the IMUs for spacecraft must maintain highly accurate inertial measurements while resisting such severe environmental factors. To meet these requirements, existing IMUs for spacecraft are large in size and expensive.

On the other hand, the gyroscopes and accelerometers developed for commercial use in-car navigation, the controllers of video games, and smartphones have become smaller and less expensive. Gyroscopes and accelerometers using Micro Electro Mechanical Systems (MEMS) technology are used for such applications. MEMS devices can be mass-produced at a low cost, using the same production process with semiconductors. Moreover, the accuracy of MEMS gyroscopes and accelerometers has been improved in recent years.

We have thus been studying ways to improve MEMS gyroscope accuracy in the severe environments space environment. [1] Studies have been also made on applying MEMS inertial sensors to IMUs for launch vehicles.[2] Along with the improved accuracy of MEMS gyroscopes and accelerometers for commercial use on ground in recent years, we have developed a low-cost and accurate Inertial Measurement Unit (IMU) using MEMS gyroscopes and accelerometers for navigation of launch vehicles and satellites, which we call as MARIN (MEMS Advanced Redundant Inertial Navigation system). Their function and accuracy have been evaluated by environment tests on the ground, experiments in radiation environments in orbit, and flight experiment during rocket launch.

This paper describes the issues of MEMS IMU for spacecraft, the design and development of MARIN, the in-orbit experiment results of MARIN on RApid Innovative payload demonstration SatellitE-2 (RAISE-2), and the flight experiment results during the launch phase on Epsilon rocket F6 and H3 rocket TF1.

2 ISSUES OF MEMS IMU FOR SPACECRAFT

Since IMUs for spacecraft need to be resistant to severe environments during launch and in orbit and provide accurate inertial measurements, the followings are the technical issues for IMUs used on spacecraft.[3]

(1) Highly accurate inertial measurements for launch guidance in high dynamics of more than 10 G
(2) Maintaining accuracy for wide temperature ranges
(3) Maintaining accuracy for high vibration and shock
(4) Ensuring normal operation in severe radiation environments

Highly accurate inertial measurements of IMU are necessary for launch vehicles to perform launch guidance for orbit insertion. Typical requirements for the inertial measurements for launch vehicles are as follows:

(a) Gyro bias error is less than about 0.1 °/h
(b) Accelerometer bias error is less than 130 μ G

Given the recent performance improvements made in MEMS gyroscopes and accelerometers for commercial use on the ground, MEMS IMU using such highly accurate MEMS inertial sensors can achieve the gyro and accelerometer bias accuracy requirements under ideal static conditions. And with special component design for severe environments for launch vehicles, such MEMS IMU will soon be available for the launch guidance of small rockets with a short mission time or for an integrated navigation system with GNSS receivers.[4]

The second issue is that wide temperature ranges cause large bias errors in inertial measurements. IMUs for spacecraft must maintain the accuracy of inertial measurements over a wide temperature range from -15 ºC to +55 ºC. The issue of maintaining accuracy over a wide temperature range can be solved using the temperature correction technique, in which inertial measurements are corrected by a polynomial approximation whose coefficients are measured in thermal tests of individual MEMS IMU. Even with the temperature correction technique, the accuracy of inertial measurements is degraded slightly compared with those under ideal static conditions. Thus, MEMS IMUs must consider the duration of inertial navigation calculations or compensate for accumulated errors in inertial navigation by other navigation sensors.

The third issue is the large bias errors in inertial measurements caused by high vibration and shock. Figures 1 and 2 show the typical vibration and shock requirements of spacecraft. This issue of maintaining accuracy for high vibration and shock is severe because MEMS gyroscopes and accelerometers have a mechanical structure, and an anti-vibration mount cannot be used for MEMS IMU due to its small size and lightweight requirements. To prevent a large bias error in inertial measurements caused by high vibration and shock, it is particularly important to design preventing structural resonances between inertial sensors and the IMU structure. We did many vibration tests and shock tests to understand this issue and applied many methods to overcome the issue. [3] Finally, we developed a special sensor mount and gyro installation method for MEMS IMU to prevent increasing large bias errors of inertial measurements caused by high vibration and shock.

The fourth issue is ensuring normal operation in severe radiation in orbit. There are two kinds of radiation effects in orbit. One is the single event effect (SEE) which occurs at the insertion of a charged cosmic ray particle and triggers instantaneous errors in electronic parts. The other is the total ionizing dose (TID) which causes accumulated damage to parts or materials by cosmic radiation. Since their mission time is short, only SEE is an issue for launch vehicle avionics equipment. As for satellites or orbital vehicles, TID is also an issue. The typical requirement of SEE for launch vehicles is that the Linear Energy Transfer (LET) threshold for parts used on the launch vehicle's equipment should be more than 31 Mev·cm2/mg. For the cost reduction of avionics equipment for spacecraft, we have tried to use automotive electronic parts which are less expensive and have environmental resistance to high vibration and over a wide temperature range. In contrast, only the space radiation environment is a major issue for electronic parts selection of spacecraft avionics equipment. Thus Japan Aerospace Exploration Agency (JAXA) has been studying radiation mitigation techniques using special electronic circuit technology for non-space electronic parts which are less expensive but do not have radiation tolerance. Radiation mitigation techniques include redundant electronic circuits, redundant modules, comparison of redundant sensors outputs, current monitor and shut-down, and module reset and recovery operation using the redundancy.[4] Based on these radiation mitigation methods, we developed a low-cost Redundant MEMS IMU "MARIN" for spacecraft using redundant CPUs and redundant MEMS gyroscopes and accelerometers.

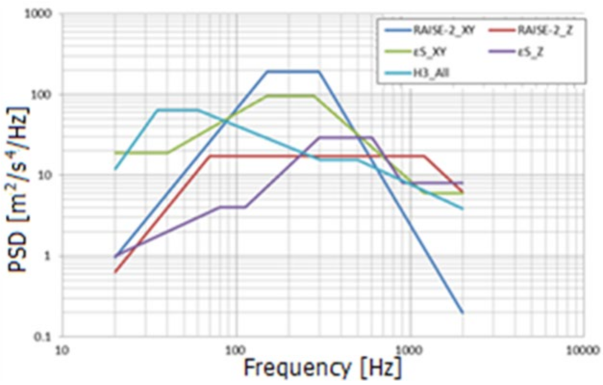


Figure 1. Random vibration test level for IMU for a launch vehicle

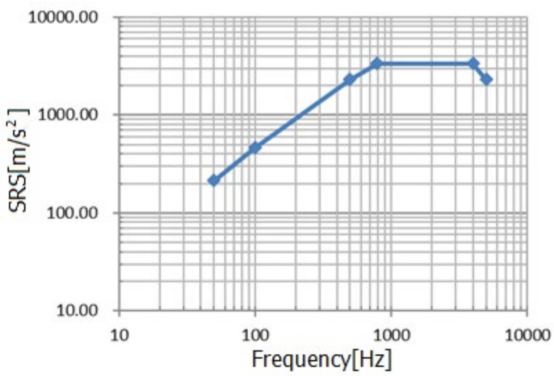


Figure 2. Shock test level for IMU for a launch vehicle

3 DESIGN AND DEVELOPMENT OF REDUNDANT MEMS IMU (MARIN)

To evaluate the accuracy and environmental resistance in the IMU configuration, we developed a redundant MEMS IMU named MARIN (MEMS Advanced Redundant Inertial Navigation system). MARIN is characterized as follows.

(1)Using high-accuracy MEMS gyroscopes and MEMS accelerometers
(2)Two redundant IMU units configured with mutual data transfer
(3)Two redundant MPUs in each IMU unit with mutual monitoring of MPU calculation
(4) One fault operative system as IMU
(5) Low-cost IMU using non-space electronic parts with radiation mitigation techniques including redundant electronic circuits, redundant modules, and current monitoring

To prevent a large bias error in inertial measurements caused by high vibration and shock, MARIN uses a special sensor mount and gyro installation method.

To cope with this radiation tolerant issue, we use the redundant MEMS IMU system as shown in Fig. 3. The basic structure is a double redundant system of MEMS IMU units. Each MEMS IMU unit has orthogonally located three gyroscopes and three accelerometers. It has two MPUs which are maturely checked for the consistency of the calculation results. There is a cross-communication link between two MEMS IMU units to check the consistency of the inertial measurements between independent units. The output data from Unit 1 contains IMU data from both Unit 1 and Unit 2. Also, the output data from Unit 2 contains IMU data from both Unit 1 and Unit 2.

The specifications of MARIN are shown in Table 1. MARIN's engineering models (EM) were developed, and Qualification Tests (QT) were conducted using MARIN EMs. After QT, three engineering fight models (EFM) were manufactured for on-orbit experiment by RAISE-2 and flight experiments during the launch phase by Epsilon rocket F6 and H3 rocket TF1. A photograph of MARIN EFM is shown in Fig. 4.

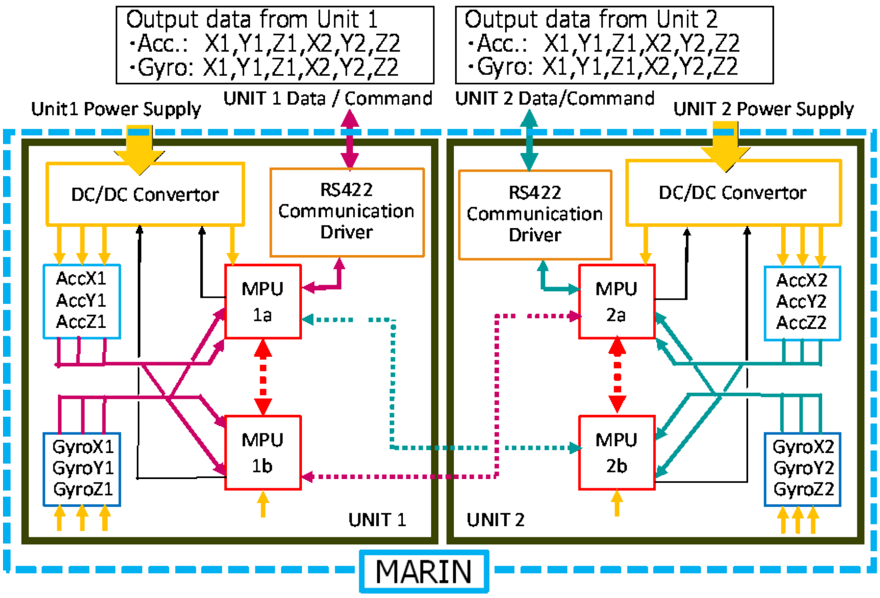


Figure 3. System Block of Redundant MEMS IMU (MARIN)

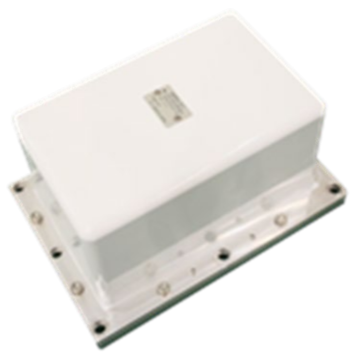


Figure 4 Photograph of Redundant MEMS IMU (MARIN)

Table 1. Specifications of Redundant MEMS IMU (MARIN)

Items		Specifications
Weight		1.24 +0.06/-0.09 (kg)
Size		152[W] × 105[L] × 80[H] mm)
Power Consumption		Less than 5W
Angle Increment Measurement		
	Range	±400deg/s
	Bias Instability	less than 0.2deg/h (Max)
Velocity Increment Measurement		
	Range	±30G
	Bias Instability	less than: 0.09 mg(Max)
Output Cycle		100Hz
Power Supply		12VDC±1V
Synchronous Signal		1PPS

4 MARIN QUALIFICATION TESTS (QT)

To evaluate MARIN's functions and performances on severe satellite and launch vehicle dynamics and environments, we did QT of MARIN for both satellite and launch vehicle on ground and radiation tests to evaluate the radiation tolerant performance of MARIN. Table 2 shows the results of QT and radiation tests. MARIN passed all checks of QT. Figures 5 and 6 are photographs of the qualification tests. The Allan variances of rate and acceleration calculated in the QT of MARIN are shown in Figs. 7 and 8, which show that the bias instabilities of rate and acceleration satisfy the bias instability requirements of MARIN shown in Table 1.

To evaluates the radiation tolerant performance of MARIN, three types of radiation test shown in Table 2 were conducted for MARIN. In the radiation tests, the circuit boards of MARIN, which are the MPU board, control board, gyro board, and power supply board, were irradiated to evaluate radiation errors on the boards in the real operation mode of MARIN. Table 3 shows the results of the proton SEE test of MARIN. Wibble parameters were estimated and the SEE error probability of MARIN was calculated by CREME 96 as follows. No Single Event Latch-up (SEL) was observed during the SEE tests.

(a)SEU error rate (using peak flux): 0.023/day
(b)SEU error rate (using average flux): 0.001/day

Figure 7 shows the TID test results for MARIN. The current of Marin increased by around 200 Gy (20 krad) and TID caused an error of around 200 Gy (20 krad).

Table 2 Qualification Tests and Radiation Test of MARIN

	Test	Content	Status
Qualification Test			
(1)	Initial Function and Performance Test	Angle increment measurement test, Velocity increment measurement test	Pass
(2)	Thermal Cycle Test	-24 °C to 65 °C , 8 cycles	Pass
(3)	Electromagnetic Compatibility Test	CS114, RE102, RS103, MIL-STD-461C, MIL-STD-461G	Pass
(4)	Vibration Test	Sinusoidal vibration test, Random vibration test requirements shown in Fig. 1	Pass
(5)	Shock Test	Low-frequency vibration test, High-frequency shock test whose requirements is shown in Fig. 2	Pass
(6)	Thermal Vacuum Test	-24 °C to 65 °C, 8 cycles, Continuous operation from atmospheric pressure to high vacuum	Pass
(7)	Final Function and Performance Test	Angle increment measurement test, Velocity increment measurement test	Pass
(8)	Thermal Shock Test	-10 °C to 60 °C, 2hour/1cycle, 100cycles,	Pass
(9)	Temperature and Humidity Test	Temperature: -10 °C to 65 °C, Humidity: 80% to 96%	Pass
Radiation Test			
(1)	Proton Single Event Effect Test	Irradiation of MARIN board modules	Acquisition of data
(2)	Heavy Ion Single Event Effect Test	Irradiation of MARIN board modules	
(3)	Total Ionizing Dose Test	Irradiation of MARIN board modules	

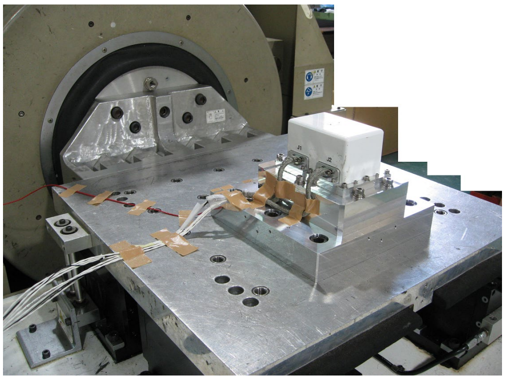


Figure 5 Vibration Test

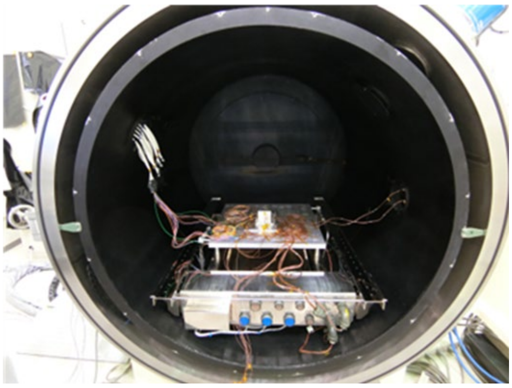


Figure 6 Thermal Vacuum Test

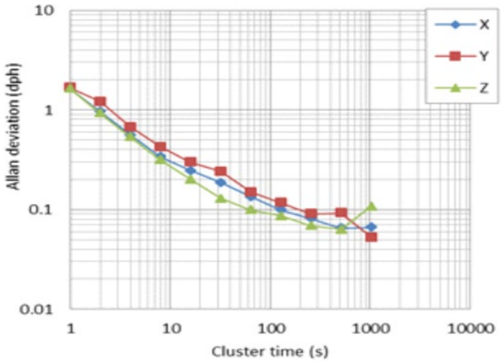


Figure 5 Allan Variance of Rate

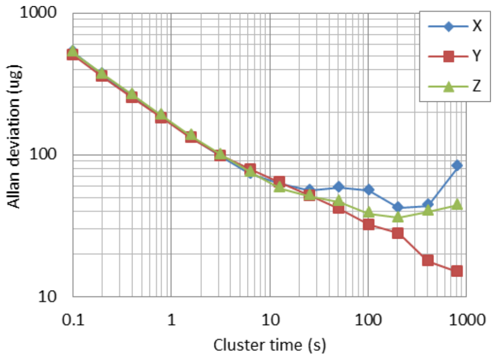


Figure 6 Allan Variance of Acceleration

Table 3 Proton Single Event Test Results

Energy (Mev)	Irradiation Time (s)	FLUX ($p/cm^2/s$)	Number of SEU	Reaction cross section (cm^2)
10	1020	1.02×10^{7}	0	0
30	900	1.16×10^{7}	1	0.96×10^{-10}
50	840	1.26×10^{7}	0	0
70	875	1.21×10^{7}	2	1.89×10^{-10}

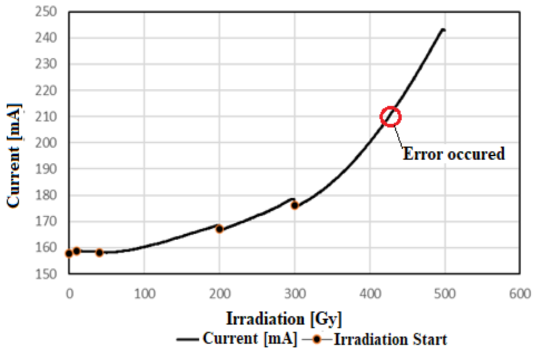


Figure 7 Total Ionising Dose Test Result

5 IN-ORBIT EXPERIMENT OF MARIN

To evaluate the radiation error rate of MARIN and demonstrate its continuous correct inertial measurement output, we did the in-orbit experiment of MARIN on RApid Innovative payload demonstration SatellitE-2 (RAISE-2) which was launched on November 22, 2021. The MARIN installed on RAISE-2 has operated normally in orbit for over a year.

5.1 Purpose and experiment plan of the in-orbit experiment of MARIN on RAISE-2

The purposes of the on-orbit experiment of MARIN on RAISE-2 are as follows.

(1) Evaluation of the SEE error rate of MARIN in-orbit
(2) Evaluation of the radiation error tolerance by redundant operation of MARIN in-orbit.
(3) Evaluation of the accuracy of inertial measurements of MARIN in-orbit
  (a) Bias instability of rate measurement
  (b) Bias instability of acceleration measurement
  (c) Angle error on attitude calculation using MARIN inertial measurements

The in-orbit experiment plan of MARIN on RAISE-2 is as follows.

1)Low-rate data acquisition of MARIN
MARIN’s low-rate (1Hz) data are acquired and transmitted to the ground during all operation periods of RAISE-2. Since the operation cycle of MARIN is 100 Hz and the transmitted cycle is 1 Hz, MARIN has an error counter counting error by 100 Hz, and the number of error occurrences can be calculated by using the error counter of 1 Hz data.

2)High-rate data acquisition during attitude maneuver
MARIN’s high-rate (10Hz) data are acquired and transmitted to the ground during the attitude maneuver which is performed within one orbital period. The attitude maneuver is executed by the following sub-maneuver sequentially.

(a1)Plus roll maneuver: roll angle is changed from 0° to 10° by 0.5°/s
(a2)Minus roll maneuver: roll angle is changed from 10° to 0° by -0.5°/s
(b1)Plus pitch maneuver: roll angle is changed from 0° to 10° by 0.5°/s
(b2)Minus pitch maneuver: roll angle is changed from 10° to 0° by -0.5°/s
(c1)Plus yaw maneuver: roll angle is changed from 0° to 10° by 0.5°/s
(c2)Minus yaw maneuver: roll angle is changed from 10° to 0° by -0.5°/s

5.2 Results of the in-orbit experiment of MARIN on RAISE-2

5.2.1 Evaluation of radiation tolerance of MARIN on RAISE-2

We counted and categorized the SEE errors of MARIN installed on RAISE-2. Table 4 shows the SEE error results and the estimated error rate. The Single Event Upset (SEU) errors were detected by the self-diagnosis function of MARIN and properly set the error flag for observed data with SEU error. Since this is a one-shot error, it did not affect the consecutive inertial measurements. At the SEU error occurrence on one IMU unit of MARIN, although the data from the IMU unit with SEU error was marked as erroneous data by an error flag, the data from the other IMU unit of MARIN measured and outputted correct inertial measurement data. One example of these sequences at a SEU error occurrence obtained on MARIN installed on RAISE-2 is shown in Fig. 8. MARIN detected an SEU error on Unit 2 Gyro (x-axis) and set an error flag on Unit1 Gyro (x-axis) error status as shown in Fig. 8, whereas Unit1 Gyro (x-axis) had no error flag and outputted correct inertial measurement data as shown in Fig.8.

The Single Event Function Interruption (SEFI) on MARIN occurred twice during the RAISE-2's operation duration which is 426 days. In the occurrence of SEFI on one IMU unit of MARIN, although the IMU unit with SEFI stopped inertial measurement, the other IMU unit of MARIN measured and outputted correct inertial measurement data. After the acquisition of the information of the SEFI, the IMU unit with SEFI was powered off and on by RAISE-2, and within a minute, MARIN returned to redundant IMU units operation. Although MARIN installed on RAISE-2 is set to be reset by an upper system, MARIN has a self-reset function when MARIN detects the function interruption.

No Single Event Latch-up (SEL) was observed during the operation duration of RAISE-2, which was expected from radiation tests.

Table 4 SEE error results of MARIN on RAISE-2

	SEU	SEFI	SEL
Total number of error in 426 days	84	2	0
Error rate	0.19 /day (once every 5 days)	0.0047/day (once every 212 days)	No errors in 426 days

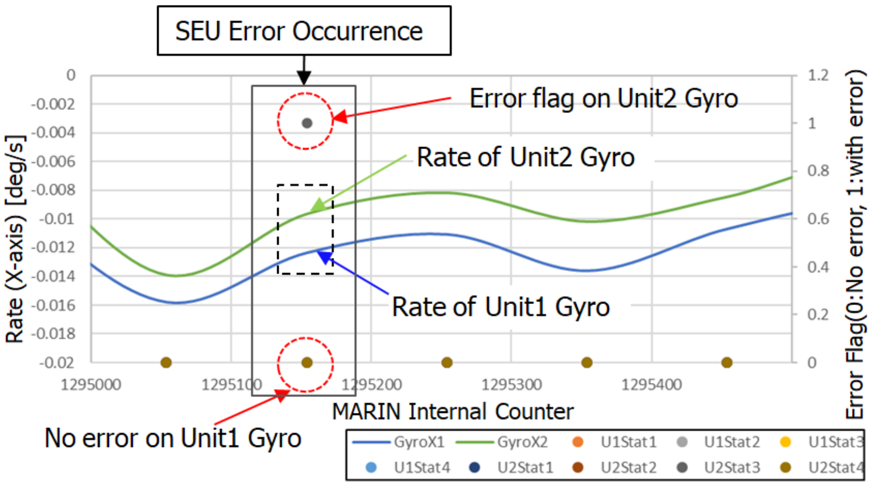


Figure 8 SEU error occurrence obtained on MARIN installed on RAISE-2

5.2.2 Evaluation of bias instability of MARIN on RAISE-2

We evaluated the bias instabilities of the rate and acceleration of MARIN obtained in orbit. Since RAISE-2 flew in an Earth-oriented attitude and its pitch rate and its control disturbance were always applied to MARIN on RAISE-2, the bias instability of rate was evaluated using the difference in rates of IMU Unit 1 and IMU Unit 2 of MARIN to compensate for attitude dynamics of RAISE-2. Figure 9 shows the bias instability of the differential rate of MARIN. Although the variance of the differential rate became larger than that of the rate itself, the bias instability of the differential rate was less than that of the rate requirement shown in Table 1. Figure 10 shows the bias instability of acceleration of MARIN IMU Units 1 and 2. The bias instability of the measured acceleration of MARIN was less than that of the rate requirement shown in Table 1.

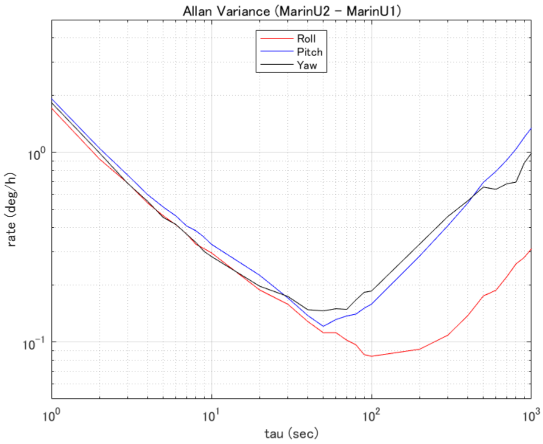


Figure 9 Allan variance of MARIN rate

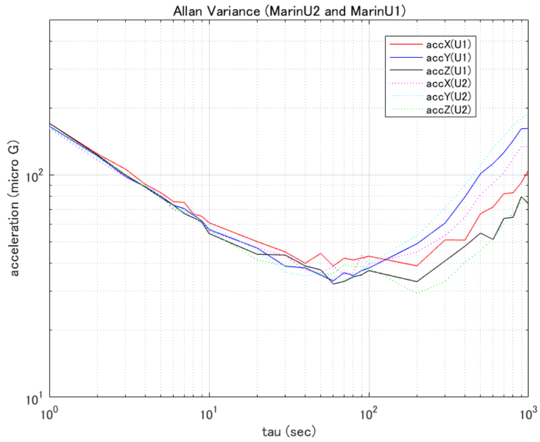


Figure 10 Allan variance of MARIN acceleration

5.2.3 Evaluation of angle error on attitude calculation using MARIN inertial measurements

The calculated attitude using angle increment data of MARIN with in-orbit alignment was compared with the estimated attitude using star tracker data of RAISE-2. In-orbit alignment was estimated as the initial attitude and bias rate of the gyroscope, comparing MARIN date and star tracker data for 400 s before starting the attitude calculation of MARIN. The evaluation of angle error on attitude calculation of MARIN was done using high-rate data, 10 Hz, during the attitude maneuver experiment.

Figure 11 compares the calculated attitudes using angle increment data of MARIN IMU Units 1 and 2 and the estimated attitude using a star tracker during the attitude maneuver experiment. The attitudes are almost the same. Figure 12 shows the differences between the calculated attitudes of MARIN and the estimated attitude using a star tracker. The attitude difference gradually increased to about 2° after 4000 s from the start of attitude calculation. The differences include attitude errors caused by MARIN inertial measurement and system errors such as the mounting error of MARIN, whose maximum is 1°.

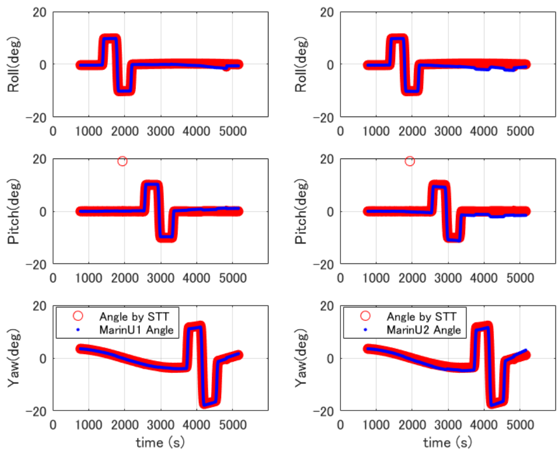


Figure 11 Calculated attitude of MARIN and estimated attitude using star tracker

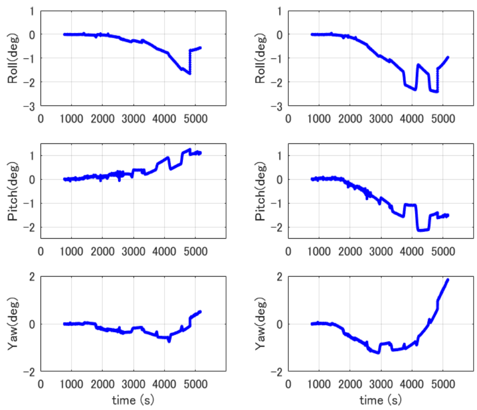


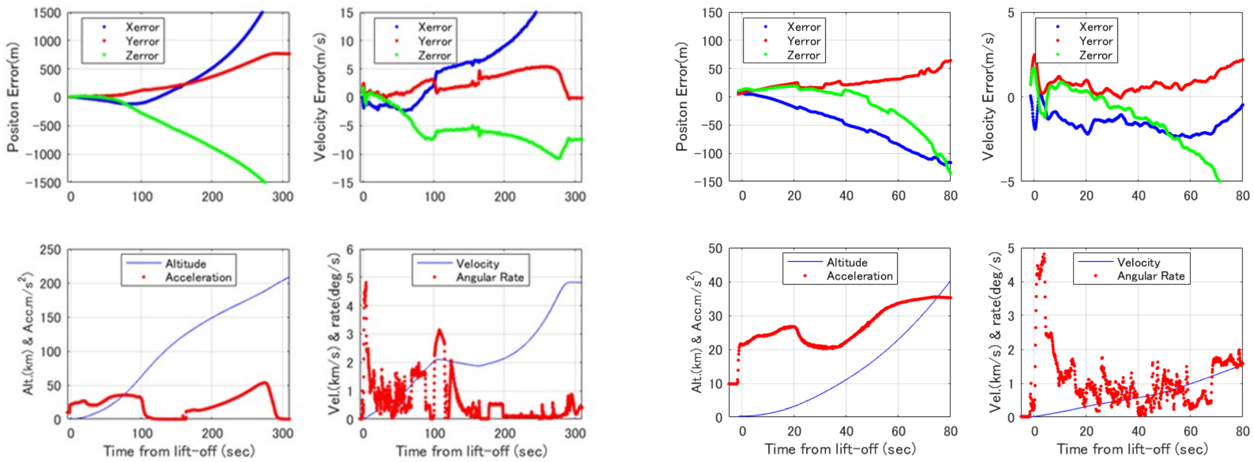
Figure 12 Difference of calculated attitude of MARIN and estimated attitude using star tracker

6 FLIGHT EXPERIMENTS OF MARIN ON LAUNCH VEHICLES

To use MARIN for the inertial measurement function of the Redundant Integrated Navigation System (RINS) [4], a flight trajectory monitoring system of launch vehicles, two MARIN flight experiments on launch vehicles were conducted. The first flight experiment on launch vehicles was done on the Epsilon rocket F6, launched on October 12, 2022. The second flight experiment was conducted on the H3 rocket TF1, launched on March 7, 2023.

6.1 Results of the flight experiment of MARIN on Epsilon rocket

MARIN was installed for the flight experiment on the second stage of Epsilon rocket F6, which was launched on October 12, 2022. MARIN operated normally during the flight. Figure 13 shows the position and velocity error in inertial navigation using MARIN data compared to rocket navigation using GPS. From Figure 13, the position and velocity errors are small first 80 s from lift-off. the velocity errors from lift-off to 5 s on the right-hand side of Figure 13 were caused by the multi-path of GPS signal on the reference rocket navigation using GPS. To avoid such a multi-path, the inertial navigation using MARIN data will be used from lift-off to the altitude of 450 m on RINS. Figure 14 shows the attitude errors of inertial navigation using MARIN data compared to the rocket attitude calculation using the rocket IMU which uses ring laser gyroscopes. The attitude errors were less than 1.5° at 300 s from lift-off.



< Lift-off ~ Second Engine Burn-out > < Around Lift-off >

Figure 13 Position and velocity error of inertial navigation using MARIN data during Epsilon F6

6.2 Results of the flight experiment of MARIN on H3 launch vehicle

MARIN was installed on the second stage of the H3 rocket TF1, and the flight experiment was conducted on the H3 rocket, \which was launched on March 7, 2023. MARIN operated normally during the flight. Figure 15 shows the position and velocity error in inertial navigation using MARIN data compared to rocket navigation using GPS. From Figure 15, the position and velocity errors are small first 80 s from lift-off. Figure 16 shows the attitude errors of inertial navigation using MARIN data compared to the rocket attitude calculation using the rocket IMU which uses ring laser gyroscopes. The attitude errors were less than 2° at 800 s from lift-off.

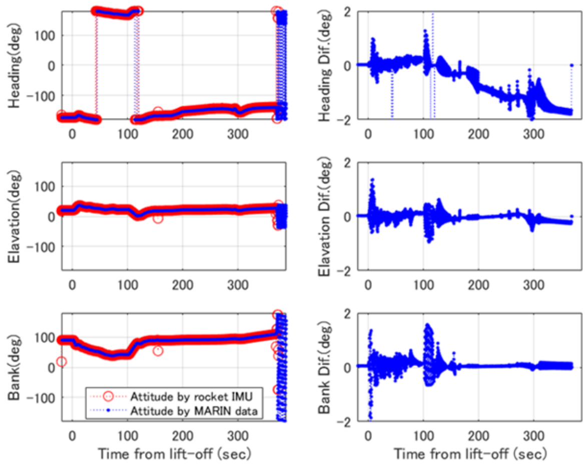


Figure 14 Attitude error of inertial navigation using MARIN data during Epsilon F6 fight

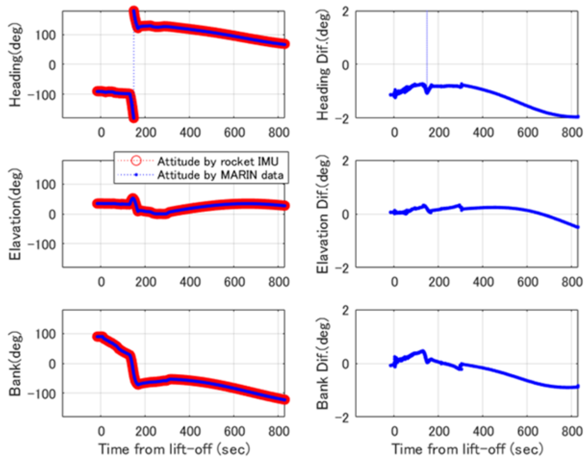
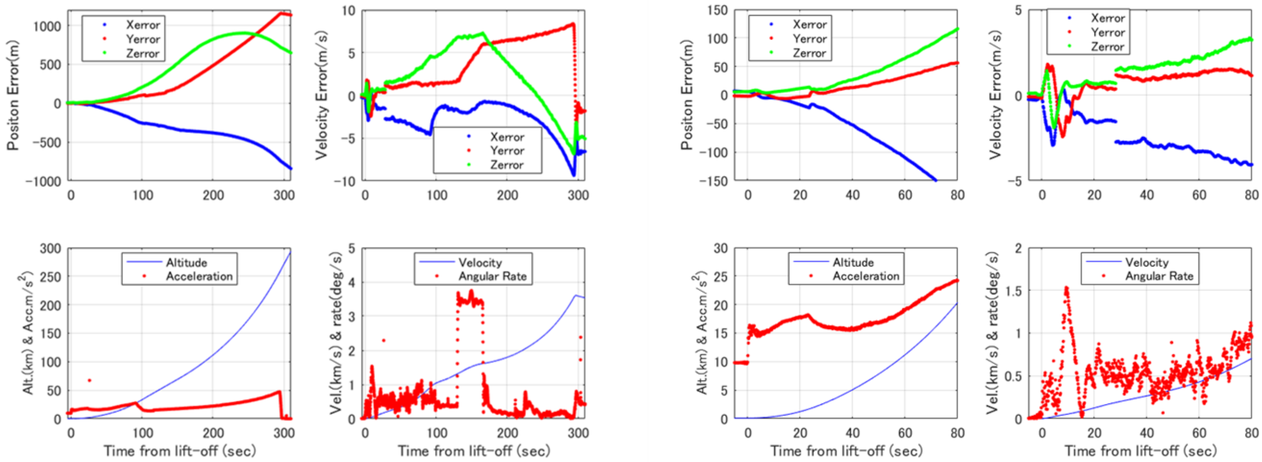


Figure 16 Attitude error of inertial navigation using MARIN data during H3 TF1 flight



< Lift-off ~ First Engine Cut-off > < Around Lift-off >

Figure 15 Position and velocity error of inertial navigation using MARIN data during H3 TF1 flight

7 CONCLUSIONS

To realize a low-cost Inertial Measurement Unit (IMU) using MEMS gyroscopes and accelerometers for navigation of spacecraft and launch vehicles, we developed MARIN, a Redundant MEMS IMU, which uses accurate MEMS gyroscopes and MEMS accelerometers, a special sensor mount, and low-cost non-space part with redundant circuit technology. MARIN passed all qualification tests, and successfully completed the in-orbit experiment on RAISE-2 and the flight experiments during the launch phase on the Epsilon rocket F6 and H3 rockets TF1.

MARIN is planned for use on Japanese flagship rockets, the H3 rocket and Epsilon S rocket, for inertial measurement for the Redundant Integrated Navigation System (RINS). We will also try to apply MARIN to satellites, lunar/planetary explorers, and orbital vehicles. MRN-01 is the product name of MARIN.

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