INDEX: A Piggy-Back Satellite for Advanced Technology Demonstration

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Abstract

This paper describes outline of the piggy-back satellite "INDEX" for demonstration of advanced satellite technologies as well as for a small scale science mission. INDEX satellite will be launched in 2002 by Japanese H2-A. The satellite is mainly controlled by the high-speed, fault-tolerant on-board RICS processor (three-voting system of SH-3). The attitude control is a compact system of three-axis stabilization. Although the size of INDEX is small (50Kg class), several newly-developed technologies are applied to the satellite system, including silicon-on-insulator devices, variable emittance radiator, solar-concentrated paddles, lithium-ion battery, and GPS receiver with all-sky antenna-coverage. These technology developments will be applied to Japanese scientific space exploration in future.

Introduction

Institute of Space and Astronautical Science (ISAS), Japan have launched a series of scientific satellites including planetary spacecraft as well as astronomical observation satellites. Although the missions have achieved fruitful scientific results, these satellites cost nearly $120 million including our own M-V launch vehicle and take longer than 6 years to be developed.

In addition to these "big, expensive, slow" missions, the Institute plans to launch "small, inexpensive, fast" piggy-back satellites as good tools to demonstrate new advanced spacecraft technology and perform science observation. The piggy-back satellite INDEX (Innovative Technology Demonstration Experiment) is planed to launch in 2002 by Japanese H2-A vehicle. This paper summarizes the outline of INDEX satellite and the advanced technology that will be demonstrated in INDEX satellite.

Outline of INDEX

INDEX will be launched as a 50kg-class piggy-back by Japanese H2-A rocket in 2002. The maximum size is 50x50x50cm3 with four solar-concentrated deployable paddles. The maximum power is about 100 W. Table 1 indicates the summary of INDEX satellite. The cost of INDEX satellite is about 4 million dollars including the dedicated ground station, and the period of

Size	$50\times50\times50$ cm^3
Mass	50 kg
Mission	10kg Science Payload will be selected in Oct., 1999 Engineering Technology Demonstration (9 items)
Power	100W, Solar-Concentrated, Deployable Solar Panels 10.5AH Lithium Manganate-Ion Battery
Attitude	Bias-Momentum Three-Axis Control Sun-Pointing, Accuracy of 0.5 degree
Launch	H2-A Vehicle in 2002
Life	Longer Than Three Months

Table 1 Outline of INDEX Satellite

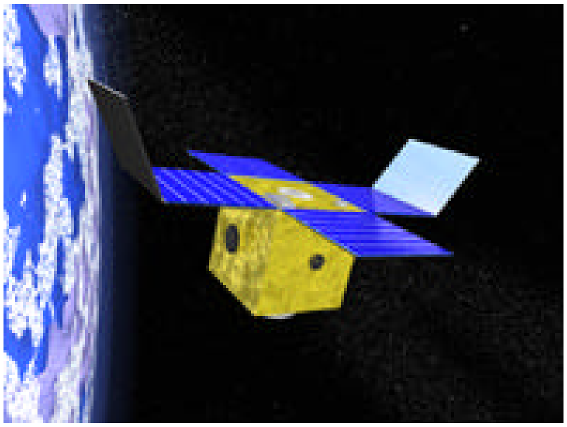

Fig.1 Image of INDEX Satellite in Orbit

development is about 2 years. Figure 1 indicates an image of INDEX satellite in orbit.

The orbit will be determined by the main mission of the H2-A vehicle in 2002. The planned orbit is sun-synchronized orbit with altitude of 690 km. The local time of the orbit is 10:30. INDEX satellite is three-axis control with bias momentum. The pointing direction is nominally toward the sun. It is possible to orient the bias momentum axis to any direction within 20 degree away from the sun as long as the power balance is satisfied. Figure 2 shows the outlook of INDEX satellite.

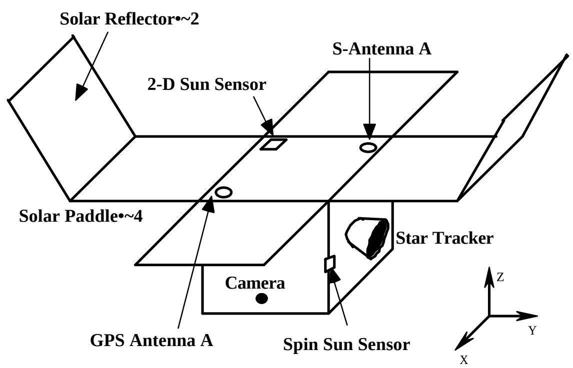

Fig.2 Outlook of INDEX Satellite

Mission Objectives

INDEX satellite has a scientific mission as well as the engineering mission. A scientific mission will be selected in October 1999 by the board of space science in ISAS. Mass of about 10 kg is allocated for science mission. Candidates of the science payload would come from space plasma research or solar physics. Space astronomy requires a large aperture to detect faint objects with higher resolution and would not be suitable for small satellite missions

ISAS has been developing the advanced spacecraft technology for future scientific satellites in STRAIGHT (Study on Reduction of Advanced Instrument Weight) program. The engineering mission is to demonstrate the advanced spacecraft technology in orbit. The advanced technologies to be tested in INDEX satellite are ;

light three-axis attitude control system
integrated spacecraft-controller based on high-speed 32bit RISC processor (SH-3, 60MIPS, 0.5W) with three voting system
lithium ion rechargeable battery using lithium manganate as main satellite battery
solar-concentrated panel with multi-junction cell
variable emittance radiator with material of perovskite.
silicon-on-insulator (SOI) technology
a GPS receiver with all-sky antenna-coverage
compact camera of CMOS active pixcell sensor
compact image-compressor using ball grid array packages
simple satellite-operation with PC terminals.

Integrated Controller Unit (ICU)

INDEX satellite is controlled by the Integrated Controller Unit (ICU). ICU manages peripheral instruments including the sensors and the actuators. Figure 3 summarizes the system block diagram. ICU consists of high speed RISC processors (Hitachi SH-3, max 60MIPS, 0.5 W) embedding DRAM memory (16MB with EDAC), system ROM (128kB with EDAC), and user ROM (384kB).

This processor is selected from advanced commertial products and is not radiation-hardening. The LET threshold of single event upset is measured to be 8MeV/(mg/cm2). SH-3 is expected to experience single event upsets in the orbit. The processor embedding DRAM memory is a fault-tolerant, three-voting system. Every time when the data goes out from the processors, the results from the three cells are compared with each other. If the result from one cell (cell A) is not consistent with other two cells (cell B and C) due to single event upset, all the data in cell A is copied from cell B through direct memory access. This reconfiguration of the processors takes less than 1 second. During this period the processors do not respond to the peripheral instruments. It is notable that this reconfiguration process is acceptable to the INDEX satellite system. LET of single event latch-up for SH-3 processor is measured to be higher than 40MeV/(mg/cm2). Protection circuit to prevent latch-up over-current is provided in the each processor cell.

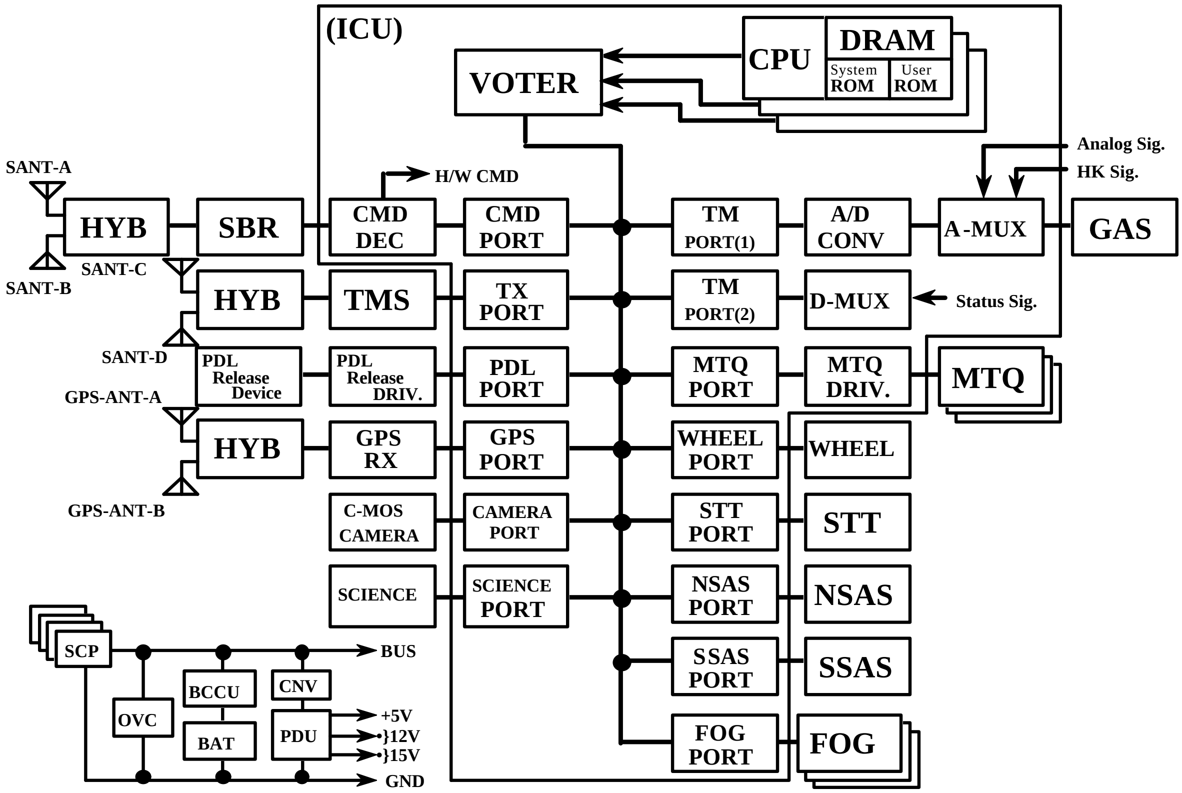

Fig.3 System Block Diagram of INDEX

Outline of ICU Software

Since the processor of ICU is a fault-tolerant and reliable system, most of the satellite functions are performed by software. Hardware is minimized to reduce mass of the satellite. All the onboard programs are stored in the user ROM region. The onboard programs are divided into the survival mode (ROM mode) and the steady-state mode (RAM mode).

The program of the survival mode is the minimum amount of the program that is necessary at least to survive the satellite, when the satellite is at the initial acquisition phase, and the emergency action such as attitude anomaly and bus under-voltage. This mode is executed on the ROM region, although limited region of RAM region is used as working memory area. The survival mode of program consists of the HK/status collection task for minimum items, the telemetry & data recorder task for minimum HK/status data, the command reception and the command execution for real-time command (not stored command), the attitude control task from the initial attitude acquisition to the Sun-pointing control at the spinning mode (not including three-axis mode), and the task to rewrite programs in the RAM region. The program of this survival mode in the ROM region can not be rewritten in orbit, since this mode is the minimum and the last controller of the satellite. This survival mode of program will be thoroughly tested and debugged at ground.

The program of steady-state mode includes the extended tasks for all satellite functions at the steady-state. This mode includes the attitude control task of three-axis stabilization, science mission task, and the stored-command task. The program of this steady-state mode is executed on the RAM region. When the program is required to be rewritten in orbit, the CPU mode returns from the steady-state mode to the survival mode (ROM mode) to execute the task of rewriting programs in the RAM region.

The SH-3 processor is provided with a simple operational system (HI-SH7). Software is written by means of C language. It is effective to develop the software by commertial environments of software developments.

Software Tasks

Software tasks are shown in Fig.4. Each task consists of the survival mode (ROM mode) and the steady-state (RAM mode), respectively. All tasks except for the command reception task repeat periodically at the rate of 8 Hz. The processor is operated at the internal clock of 32 MHz. This capability is enough to perform the tasks of INDEX satellite control. Each task is allocated the fixed time frame, which is long enough for the task to finish the job. The command reception task is initiated as the hardware interrupt, which is generated by the command decoder circuit.

When a command from the ground station is received through the command-decoder circuit, the hardware interrupt to the processor occurs. Then the processor ceases the present task to read the command message and executes it in case of real time command. Stored command is processed at the designated timing by the programmed-command task.

The telemetry & data recorder task manages the telemetry processing, and control of the data recorder in the ICU. All the real-time data and the stored-data are accumulated in the RAM region of ICU. There are several format-modes of telemetry data. One of these modes is selected depending on the satellite operation and the telemetry bit rate. The telemetry frames are generated by the software. The S band transmitter is connected to the data bus of the ICU through the telemetry buffer circuit. The telemetry & data recorder task manages to transfer the new telemetry data to the telemetry buffer circuit at the rate of ICU bus speed (3 Mbps).

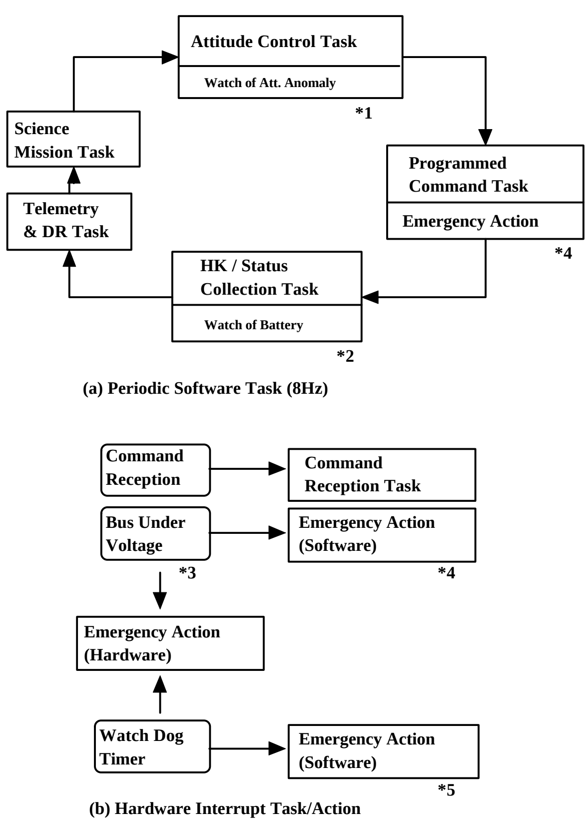

Fig.4 Software Tasks of ICU

The HK/status collection task manages to collect House-Keeping (HK) data and the status data through the ICU bus. The HK/status collection task keeps calculating the depth-of-discharge of the bus battery by integrating discharge current. If the battery is found to over-discharge, the ICU initiates the emergency action.

The attitude control task manages all functions of the attitude control, including data collection of the attitude sensors, calculation of the control algorithm, and the control of the attitude actuators. The details are described in the next section. If attitude anomaly is found by this task, the ICU initiates the emergency action.

Attitude Control System

The attitude control system of INDEX satellite is bias-momentum three-axis stabilized system. Various missions of space science such as imaging observation and star acquisition require three-axis stabilized system. The bias-momentum stabilization can achieve about 1 degree accuracy of attitude pointing. In case that even higher accuracy and high speed of the attitude maneuver are required, it is relatively easy to extend the system to zero-momentum stabilized system by adding another two momentum wheels. Also it is easy to change the system to spin stabilized system by removing the wheel and the sensors for the three-axis control.

The sensors for attitude control are a spin type sun sensor (SSAS), two-dimensional sun sensor (NSAS), a star tracker (STT), three-axis geomagnetic aspect sensor (GAS, magnetometer), and three-axis optical fiber gyroscopes (FOG). The star tracker can determine attitude of the satellite with accuracy of 0.05 degree, when the earth and the sun are outside of forbidden angle.

When the star tracker does not work, the two-dimensional sun sensor and the fiber optical gyroscopes can determine the attitude with accuracy of 0.5 degree. The actuators of the attitude control are a small momentum wheel (WH) which provides the satellite with bias-momentum (1Nms), and three-axis magnetic torquers (MTQ). Table 2 shows outline of the sensors and the actuators.

System	Bias-Momentum Three-Axis Control	
Pointing	Sun-Pointing, off-set±20deg	
Accuracy	0.5degree	
Actuators and Sensors	Wheel	1 axis, 2.2kg, 1.5Nms, 0.01Nm
	Torquer	3 axis, 0.2kg×3, $2Am^2$
	Magnetometer	3 axis, 0.5kg
	Star Tracker	1.3kg, 0.05deg accuracy
	Non-Spin SAS	1.0kg, 0.05deg accuracy
	Spin-SAS	0.1kg, 0.5deg accuracy
	Optical Gyro	3 axis, 0.15kg×3, 0.5deg/h drift

Table 2 Outline of Attitude Control Subsystem

The control algorithm in the three-axis pointing phase is based on the compact control law that was applied to our astronomy satellite ASTRO-C in 1987[1]. This law is based on a linear feed-back of attitude error and rate, and achieves both angular momentum control and nutation damping simultaneously based on a simple control law.

The target attitude is inertially fixed and is represented by O-XYZ coordinate system in Fig.5 (a). The body-fixed system is depicted as O-xyz, where each axis coinsides with the principal moment of inertia axis. The satellite has a momentum wheel whose axis is the z- axis. The relation between the inertial coordinate O-XYZ and the body-fixed coordinate O-xyz is expressed by Euler angles $_x$, $_y$, $_z$ which correspond to error angles of the satellite.

Figure 5 (b) shows a free nutation of the satellite in the $_x$- $_y$ plane. The point H is the center of the nutation and corresponds to the angular momentum vector direction. The point P denotes the z-axis of the satellite. The torque for angular momentum vector control is along HO, while the one for nutation dumping is along HP. The present control law is that the torque controls of

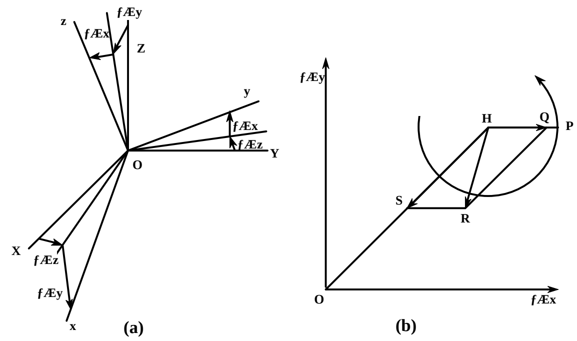

Fig.5 Principle of Pointing. (a) Target Attitude C XYZ and body-fixed coordinate O-xyz. *x*, *y*, are small Eular angles. (b) shows a free nutatio of satellite in *x*- *y* plane.

the former and the latter are simultaneously achieved. In Fig. 5 (b), HS and HQ represent the torques for momentum control and nutation damping, respectively. The torque provided by the magnetic torquers is HR, which is vector-sum of HS and HQ. The magnetic moment M of the satellite is controlled as

$$\mathbf{M} = g(\Delta\mathbf{H} \times \mathbf{B})$$

$$\Delta\mathbf{H} = \begin{pmatrix} h_0(q_y + a\dot{q}_x)/w_0 \\ -h_0(q_x - a\dot{q}_y)/w_0 \\ h_w - h_0 \end{pmatrix} \quad (1)$$

where *g* and are constants, ***B****=(Bx,By,Bz)* is the geomagnetic field, $_0$ is the nutation angular velocity , h_w is the wheel angular momentum at each instant, and ho is the nominal value of h_w.

Initial Attitude Maneuver and Satellite Operation

This section describes the outline of the attitude maneuver[2] and the satellite operation from the separation from the launch vehicle to the steady state. INDEX satellite is separated from the H2-A vehicle after the main payload satellite is separated. The separation of INDEX satellite is accomplished by spring release and undesired attitude rate of about 5 rpm is provided. The direction of the initial angular momentum is not clarified in the launch vehicle interface. The requirement of piggy-back launch of H2-A imposes that the main power of INDEX satellite is off before the separation.

Fig. 6 shows the flow of the initial attitude maneuver and the satellite operation. All processes from the separation to the spinning,

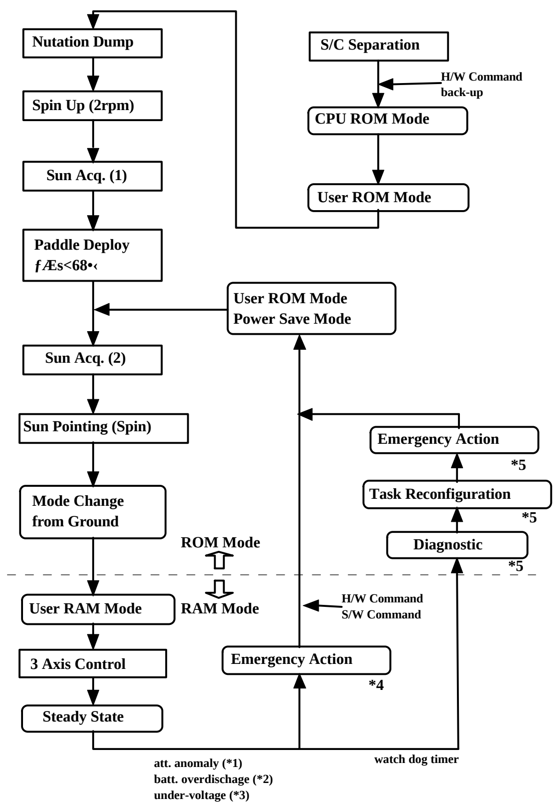

Fig.6 Sequence of Attitude Maneuver and Satellite Operation for Initial Acquisition, Steady State, and Emergency Action

sun-pointing mode are autonomously executed in orbit without any assist from ground.

The mechanical switch is closed at the separation and main power is supplied from the satellite battery to the satellite bus line. The processor in the ICU performs the initial check for the CPUs, the memory and the voting function with three CPU cells. Then the CPUs execute the programs of survival mode mainly on the ROM region.

The first stage of the attitude maneuver is the nutation dumping process by means of the magnetometer and the magnetic torquers. The nutation motion induces the temporal change rate of the magnetic field in the satellite body frame. The control law of the active nutation control is written as

$$M_z = - M_0 sign\ (d\ B_z/dt\ ), \qquad (2)$$

where d B_z/dt is the temporal change rate of the magnetic field in the body-fix coordinate, and M_0 is the magnetic moment of the torquer. For sake of hardware simplicity, the torquer control is a bang-bang system. The samplings of the geomagnetic measurement are performed at two timings to determine the polarity of the torquer driving. This measurement is executed when the magnetic field produced by the torquers are steady state to avoid interference to the magnetic measurement.

The second stage of the maneuver is spinning up to the rate of 2rpm by means of the spin-type sun sensor, the magnetometer, and the magnetic torquers. The spin-type sun sensor and the two-dimensional sun sensor are switched to measure the spin rate and the sun aspect angle.

The third stage of the maneuver is the sun acquisition by means of the spin-type sun sensor, the magnetometer, and the magnetic torquers. When the spin-type sun sensor with a fan-shaped field-of-view detects sun pulse in a spin motion, the geomagnetic field is measured to determine the polarity of the torquer. The torquers are off during the measurement. The control law is written as

$$M_z = - M_0 sign\ (- B_x \quad B_y\ ), \qquad (3)$$

where B_x, B_y are values at the sun pulse. Driving of magnetic torquers is suspended during this measurement in order to avoid the magnetic interference. Once the sun-aspect angle is less than 68 degree, the solar paddles are deployed to obtain higher power than the stowed condition. The spin rate becomes slow down to 1.6rpm.

Monte Carlo simulation of the initial attitude acquisition indicates the ellapsed time from the separation to the paddle deployment is less than 4 hours. The power consumption of the bus is about 18W for the survival mode at the initial acquisition. Capacity of the bus battery is about 200 WH (DOD=75%), which can supply the bus power for 11 hours. The battery capacity is enough to supply the bus power till the solar paddles are deployed.

The first contact to the ground tracking station will be about 12 hours later after the separation from the H2-A. At this time the satellite is supposed to be in spin-stabilized sun-pointing mode. After the initial health check, command from ground changes the CPU mode from the survival mode (ROM mode) to the steady-state mode (RAM mode) in which the program is executed in the RAM region. The attitude control changes to the three-axis stabilization. At first, the satellite body is spinned up to 3.75rpm. Then the momentum wheel is switched on and acquires momentum from the satellite body. Finally the satellite is three-axis stabilized with bias-momentum of 0.75Nms.

Emergency Action

There are two kinds of emergency detection; software detection and hardware detection. The software detection consists of attitude anomaly and power anomaly. The attitude control task can detect attitude anomaly by means of the sun sensors. There are two conditions of the attitude anomaly. One is that the sun-aspect angle is measured to be larger than 40 degree (TBD) at the sun-pointing mode. Another is that the sun sensor does not detect the sun for longer than 40 minutes relating to the eclipse time.

The software detection of power anomaly is concerning depth-of-discharge for the bus battery. In general Lithium-ion battery that is used for INDEX has relatively flat characteristics of voltage v.s. depth-of-discharge. Depth-of-discharge, which is numerically calculated by integrating discharge current, is more suitable than the bus voltage. As hardware detection of under-voltage of the bus power is utilized for the sake of simplicity. It is expected to act only when software detection fails to detect the emergency. When the Power Control Unit (PCU) detects the under-voltage of the bus, hardware interrupt occurs to the CPU.

When the emergency is detected by software or hardware, the programmed-command task executes the emergency action (Fig.6). The CPU program jumps to the ROM region from the program of RAM region. Instruments which are not necessary to survive the satellite are switched off to save power. The momentum wheel is off and the satellite body starts spinning. The sun acquisition is executed by means of the spin-type sun sensor, the magnetometer, and the magnetic torquers.

Power Subsystem

Advanced technologies of solar-concentrated solar paddles and lithium-ion batteries are applied to the power subsystem. INDEX satellite has four deployable solar paddles with solar concentrators, which are newly developed for INDEX satellite. The solar cells are multi-junction cell with conversion efficiency of 26% (triple junction cell) or 30% (guard-junction cell). The power to be generated is about 100 W in sunshine. The concentrators are very light reflectors attached to the tip of the paddles with angle of 60 degree. These solar reflectors concentrate the solar intensity on the solar cells by a factor 1.4. In the stowed position the reflectors are installed between the solar paddles and the satellite body so that the reflectors do not block the solar light. The solar paddles and the reflectors are deployed in an action of deployment by a non-explosive device using shape-memory alloy.

Lithium-ion battery using lithium manganate that is newly developed is applied to INDEX satellite. The batteries with capacity of 3.5 AH are used as 3-parallel sets of 7-series batteries. Depth-of-discharge is expected to be about 10% in orbit. Cycle test has been performed in ground up to 2000 cycles. Lithium-ion battery using lithium manganate is expected to more robust against overcharge and overdischarge than conventional ones using lithium cobaltite. The charge/discharge circuit can be simpler than latter. The charging circuit of INDEX is a simple circuit of charging with a resistance load, while conventional ones using lithium cobaltite need complex circuits of constant-current / constant-voltage charging. The shadow rate of the orbit of INDEX is about 35%. The bus power is unregulated power of 30-25 V. Regulated voltages of +15V, +12V,-12V, and +5V are generated by small DC/DC converters (Lambda) with noise suppression filters. Several instruments in the INDEX satellite have possibility of single event latch-up. These instruments are provided with current breaker circuits to prevent over-current.

Communications

Communications subsystem of INDEX is inexpensive even though relatively high bit rate can be achieved. The reason for this point is that ranging function is eliminated from the communication instruments. Orbit determination depends on the onboard GPS receiver and doppler measurement. As back-up two-line-element information from Goddard Flight Center, NASA is utilized for the antenna tracking.

The frequency of communications is S-band.

The up-link and the down-link is non-coherent. The on-board receiver is a long-phase-lock-loop PM receiver and the transmitter is PM transmitter of 2W output. The antennas are low-gain, patch antennas which are dedicated for the up-link and down-link. Two sets of antennas are installed at +z surface and -z surface. They are combined by hybrid circuits to communicate with ground for any attitude of the satellite. The dedicated ground station with an antenna of 3 meter in diameter will be constructed in the main campus of ISAS. Bit rates of up-link and down-link are 1kbps and 32kbps, respectively. When the science mission requires higher bit rate for the down-link, the antenna of 20meter at Kagoshima Space Center, Japan is utilized. The baseband equipment based on PCs is connected to the 20m facility and down-link of 256 kbps can be achieved.

GPS Receiver with All-Sky Antenna-Coverage

When GPS on-board receivers can measure position of the satellite, the communication instrument need no longer the ranging function with coherent RF-link. Also operation of the range measurement can be eliminated. INDEX satellite will test the compact onboard GPS receiver based upon commercial GPS receivers for car-navigation application. The antenna with an preamplifier (NAY-3400G) is mass of 160g and the receiver including digital processing function (CCA-370HJ) is mass of 35g.

In the ground application, the range-rate between the GPS satellites and the user is less than ±1 km/sec, which results in doppler shift of ±5kHz. In this GPS receiver a frequency stability of the local oscillator is±12kHz and the frequency search range is selected to be ±17 kHz. In the space application, the doppler shift depends on the orbit of the user satellite. According to computer simulation, it is more likely than probability of 99% that four GPS satellites having range rate of less than ±4 km/s are visible from the INDEX satellite in orbit. Software of the GPS receiver is modified so that the frequency search range is extended up to ±32kHz (±20kHz±12kHz). The acquisition time of the GPS receiver for cold start is less than 5 minutes in the ground. In the orbit it is expected that the GPS receiver can execute the acquisition within 10 minutes, taking the frequency search range into account. It is likely that the GPS satellite is still visible from INDEX satellite in 10 minutes.

GPS satellites may appear at any direction in 4 steradian from the GPS receiver, taking orbit and attitude of the user satellite into account. A GPS receiver with all-sky antenna-coverage is very effective for space application. As long as knowledge of the authors is concerned however, a GPS receiver with all-sky antenna-coverage has not been applied to space. Fig.7 shows the configurations of the newly proposed system as well as the conventional systems. One of the conventional systems is switching technique of the GPS antenna by commanding (Fig.7(a)). Strictly speaking, however, this system does not have all-sky antenna-coverage at instant. Another one of the conventional systems is shown in Fig.7(b). Each antenna is connected to a GPS receiver/signal processor, and then the digital outputs from each receiver are combined at another processor to determine the position of the user satellite. The problem of this system is that the dedicated system has to be developed for space application, which needs resources of cost and mass. The newly proposed system is quite simple as shown in Fig.7(c). Two sets of the GPS antenna with the preamplifier are combined by means of the "Wilkinson power divider" circuit. The Wilkinson power divider can feed dc power from the receiver circuit to the preamplifiers. In this system, most of commercial GPS systems may be potentially applied to space GPS with all-sky antenna-coverage. It is experimentally confirmed in the ground that this system can acquire signals from the GPS satellites located in the coverages of both the antennas. Also it is confirmed that RF interference effect from the two antennas do not affect on the receiver function.

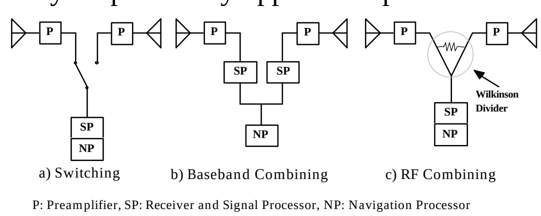

Fig.7 Configuration of GPS Receiver and Antennas.

Silicon-On-Insulator Technology

For long time radiation-hardening electronic components for space application have depended on MIL standard components. It, however, is said that many radiation-hardening components based on MIL standard are fading out. Also besides the MIL standard components, advanced electronic devices such as high-speed processors have to be applied to space technology.

ISAS is planning that the newly emerging technology of silicon-on-insulator (SOI) devices are widely applied to Japanese space projects. Many semiconductor companies in Japan are intensively developing SOI devices and soon many of them will be available in the markets. We are testing these SOI devices including total doze, single event. One example of the test is that 256kbit SOI SRAM (Honeywell, HT6256), which is not space-qualified, can survive after 1Mrad exposure. This SRAM device will be used in the INDEX satellite.

In SOI devices the electronic active layers are fabricated on the insulator layer, while in conventional bulk devices the active layers are fabricated on the silicon layer. Therefore, in case of SOI devices, when a high energy particle hits the device and generates many ionized charged-particles, only few of them flow in the electronic active layer. This is the reason why SOI devices are robust against single events. Total doze effects are closely related to depth of the gate oxide SiO2 insulator between the metal gate and the electronic active layer. We analyze the sectional structure of the SOI devices and estimate the robustness against total doze effects. SOI devices that are selected in this way will be applied to INDEX instruments. Figure 8 is an example of sectional structure of SOI device from one of Japanese manufacturers.

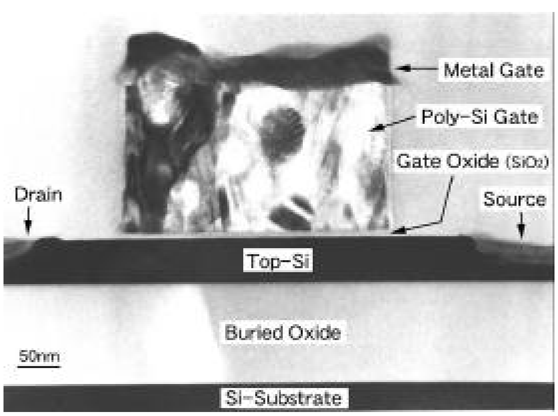

Fig.8 Sectional Structure of SOI device.

Variable Emittance Radiator with Perovskite Material

Thermal louvers are important devices for thermal control in satellites. New types of the light louver are under development. One type is the micro-louver by means of micromachine technology. Similar structure to conventional louvers is miniaturized by micro-fabrication technology. The actuator is either of bi-metal or electrostatic force. Another type is electrically variable material. When voltage of few volt is applied to the film, the thermal emittance of the film is changed.

The newly developing type of variable emittance is perovskite material ($La_{0.7}Ca_{0.3}MnO_3$ and $La_{0.7}Sr_{0.3}MnO_3$). These materials have the metal-like properties at higher temperature and the dielectric-like properties at lower temperature. The temperature of transition is dependent on the material composition and is controllable. Figure 9 shows an example of the thermal emittance v.s. temperature. Film of this material is under development for the variable emittance surface. INDEX satellite will test this type of radiators in orbit. This film will be installed on the rear side of the solar paddles to measure the paddle temperature. It is expected that thermal cycle of the paddle is mitigated between sunshine and shadow by the variable emittance film.

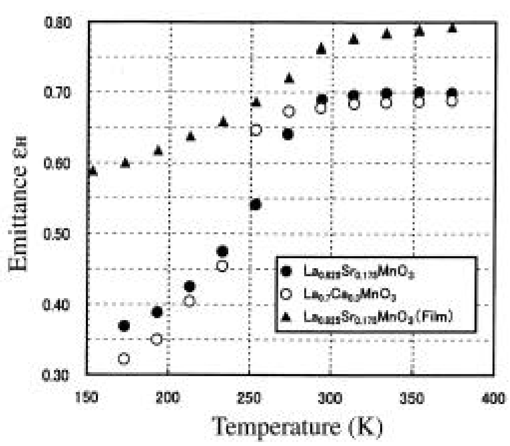

Fig.9 Thermally-Variable Emittance of perovskite material.

Conclusion

This paper describes outline of the piggy-back satellite "INDEX" for demonstration of advanced satellite technologies as well as for a small scale science mission. INDEX satellite will be launched in 2002 by Japanese H2-A. The satellite is mainly controlled by the high-speed, fault-tolerant on-board RICS processor (three-voting system of SH-3). The attitude control is a compact system of three-axis stabilization. Although the size of INDEX is small (50Kg class), several newly-developed technologies are applied to the satellite system, including silicon-on-insulator devices, variable emittance radiator, solar-concentrated paddles, lithium-ion battery, and GPS receiver with all-sky antenna-coverage. These technology developments will be applied to Japanese scientific space exploration in future.

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