THE INERTIAL STELLAR COMPASS: A MULTIFUNCTION, LOW POWER, ATTITUDE DETERMINATION TECHNOLOGY BREAKTHROUGH

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The Inertial Stellar Compass (ISC) is a miniature, low power, stellar inertial attitude determination system with an accuracy of better than 0.1º (1 sigma) in three axes. The ISC consumes only 3.5 Watts of power and is contained in a 2.5 kg package. With its embedded on-board processor, the ISC provides attitude quaternion information and has Lost-in-Space (LIS) initialization capability. The attitude accuracy and LIS capability are provided by combining a wide field of view Active Pixel Sensor (APS) star camera and Micro-ElectroMechanical System (MEMS) inertial sensor information in an integrated sensor system. The performance and small form factor make the ISC a useful sensor for a wide range of missions. In particular, the ISC represents an enabling, fully integrated, micro-satellite attitude determination system. Other applications include using the ISC as a "single sensor" solution for attitude determination on medium performance spacecraft and as a "bolt on" independent safe-hold sensor or coarse acquisition sensor for many other spacecraft.

NASA's New Millennium Program (NMP) has selected the ISC technology for a Space Technology 6 (ST6) flight validation experiment scheduled for 2004. NMP missions, such as ST6, are intended to validate advanced technologies that have not flown in space in order to reduce the risk associated with their infusion into future NASA missions.

This paper describes the design, operation, and performance of the ISC and outlines the technology validation plan. A number of mission applications for the ISC technology are highlighted, both for the baseline ST6 ISC configuration and more ambitious applications where ISC hardware and software modifications would be required. These applications demonstrate the wide range of Space and Earth Science missions that would benefit from infusion of the ISC technology.

INTRODUCTION

While the attitude needs of various spacecraft differ, the vast majority of them use a combination of sensors to determine how they are oriented in space. Traditionally, the size, weight, power, and integration requirements for many of these attitude determination sensors place a large burden on the spacecraft bus designer and noticeably reduce the amount of payload potential for small and micro-spacecraft. In order to expand the mission capabilities of these small spacecraft, the ability to determine their attitude using a low power and low mass system is of great value.

Draper is developing the ISC to answer the need for a large dynamic range, high accuracy, low power, and real-time attitude determination sensor for micro-spacecraft. The ISC, developed under NASA's New Millennium Program's Space Technology 6 Project, is an enabling technology that utilizes Draper's expertise in MEMS sensor technology, inertial navigation algorithms, and APS imagers. It integrates a wide field-of-view star camera, MEMS gyros, software, and associated processing electronics into a 2.5 kg two-piece attitude sensor that consumes less than 3.5 Watts of power.

The ISC is an innovative attitude determination sensor that combines these emerging MEMS and APS technologies into an integrated package that will be space-validated in 2004. Key ISC performance features include:

- Better than 0.1° (1-sigma) accuracy in each axis
- High-rate maneuver capability (up to 50°/sec)
- Self-initializing (over 99% of the sky)
- Low Mass ~ 2.5 kg
- Low Power ~ 3.5 W

Among the key advantages of the ISC are its ultra-low power, ease of integration with a host spacecraft, and ability to maintain a better than 0.1° accuracy during a high rate (up to 50°/sec) maneuver. As a sensor, the integrated functionality and high rate capability are unique and represent a step forward in spacecraft technology. All of the navigation algorithms and sensor fusion reside within the ISC, thereby eliminating the typical integration of the star camera data with the spacecraft attitude control software. The ISC output, consisting of an Earth Centered Inertial (ECI) attitude quaternion and attitude rates, provides a "bolt on" attitude determination solution.

SYSTEM OVERVIEW

During normal operation, attitude information is provided by the ISC's MEMS gyros. The gyros sense inertial rates that are sampled at a high frequency (320 Hz). The raw gyro data is compensated and processed through a Kalman filter to produce the output quaternion, which is transmitted to the host spacecraft in real time, at a frequency of 5 Hz. The star camera is used periodically (every few minutes) to obtain a camera quaternion that enables the gyro errors to be removed and the inherent drift of the gyros to be calibrated and

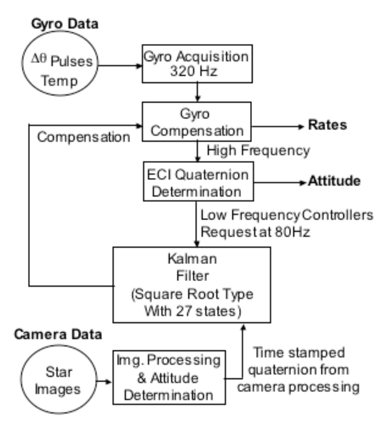


Figure 1: ISC System Data Flow

compensated. A simple system data flow is shown in Figure 1.

The fusion of star camera and MEMS gyro data provides a significant improvement in the performance of both devices. The gyros provide information continuously to the ISC during periods when the spacecraft may be undergoing high slew rates such that the camera cannot obtain good star images (due to unacceptable pixel smearing and periods when the camera is occluded by the Sun, moon, or Earth). The camera provides information for periodic updates to support gyro compensation and system initialization.

To self-initialize the ISC, star field images taken by the ISC's camera electronics are processed to provide a camera quaternion. Stars in the image are identified using a Lost-in-Space attitude determination algorithm developed by Mortari, which analyzes the image against a catalog of 1500 stars to help identify the camera's line of sight without any prior knowledge of the spacecraft's attitude.[1] Once initialized, the gyros are used to maintain attitude knowledge continuously until the next stellar update can be obtained to support gyro compensation. The complementary use of the gyros and camera data help the spacecraft overcome difficulties in providing attitude knowledge during transients, high slew rates (up to 50º/s), or periods of star camera occlusion.

ISC Performance

The ISC maintains an attitude determination accuracy of 0.1º (1 sigma) or better in each axis provided star sightings occur at least once every 150 seconds, the star sightings occur at a host spacecraft body rate <0.25º/sec, the accumulated angle between star sightings is <360º per axis, and the maximum body rate is <50º/sec per axis. The NMP ST6 top-level requirements state that accuracy will be maintained at all times using less than 4.5 Watts of power with an ISC packaged weight of less than 3 kg (excluding mission specific baffle and cabling). Current Best Estimates (CBE) for operational power consumption is 3.5 Watts and 2.5 kg for mass.

An analytical model was constructed to appropriately distribute performance requirements to subassemblies in order to capture major error sources, establish the robustness of the design, and identify the highly sensitive parameters. The overall system accuracy can be met if the camera system can achieve a 75 arcsec accuracy and the gyro subsystem achieves a 300 arcsec accuracy. This apportionment requires the MEMs gyros have an ARW (Angle Random Walk) of less than 0.16 deg/rt-hr, with a bias drift of less than 3.3. deg/hr, and scale factor errors of less than 100 ppm.

Gyro-based attitude accuracy is driven by the ability of the Kalman filter to estimate gyro misalignment, bias instability, and scale factor errors. The stability of the gyro bias in the ISC has been modeled as a first order Markov time-dependent variable with a 45-minute time constant. Gyro test data was reviewed to better estimate the bias stability as both a function of time and temperature. Results indicate that 75% of the bias stability is attributed to

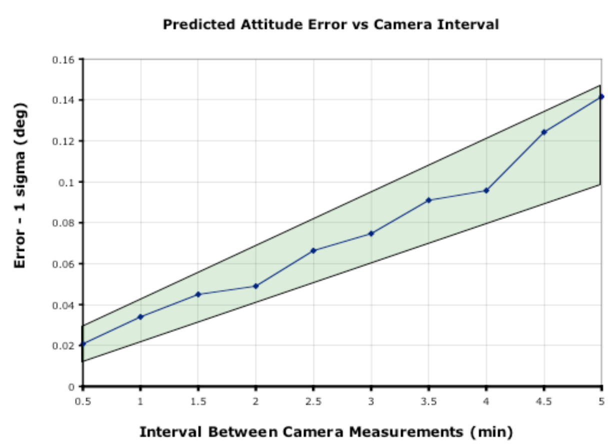


Figure 2: Attitude Error vs. Camera Interval

temperature fluctuations and 25% to the typical time-driven bias instability. The same was done for the gyro scale factor and resulted in a temperature-dependent variation of 75 ppm and a time dependent variation of 25 ppm. An expanded range of gyro thermal transients was investigated given various orbits, solar exposure, and baseplate conductivities. This analysis indicated that the mechanical design provides excellent thermal resistance between the ISC housing and the internal gyro modules. Variations of 0.09 deg-C/min at the camera board and the gyro board occurred given a 10 deg-C/hr rate of change at the ISC baseplate, which satisfies the accuracy requirements.

Camera error sources were quantified and modeled both through analysis and experiment. Pixel level effects such as detector noise, interpolation errors, and pixel smearing were individually modeled and fed into an overall Field-Of-View (FOV) level analysis in the camera model. Lens-detector misalignments, astronomical aberration, line scan aberration, and relative star distributions in a randomly chosen FOV were modeled (and verified through test) to estimate camera accuracy both in typical and worst case situations. Worst case operating conditions included a 40ºC focal plane, 0.25º/sec angular slew rate, and the maximum mechanical misalignment tolerance (1/3 pixel in pitch/yaw and 25 microns in focal length). Under these worst-case operating conditions, the camera results predict an accuracy of 55 arc-seconds in roll and 47 arc-seconds in pitch/yaw and estimate the median number of visible stars to be 13. For the worst-case analysis, the error is dominated by the 8 micron mechanical misalignment in the pitch-yaw plane. Typical conditions such as a 20ºC focal plane, tight mechanical tolerance (1/6 pixel in pitch/yaw and 10 microns in focal length), and a quasi-steady host spacecraft (0.1º/sec) result in approximately 37 arc-seconds in roll and 18 arc-seconds in pitch/yaw and predict the median number of stars to be 14. In both typical and worst case operating conditions, the percent of trials with greater than 3 stars in the FOV was greater than 99% and satisfied initialization requirements.

ISC camera data is provided as truth to update gyro states within the Kalman filter. The camera update rate is variable and ranges from 30 seconds to 20 minutes. A longer camera update interval results in decreased attitude accuracy. Figure 2 illustrates the predicted accuracies of the ISC as a function of camera update rate. There is approximately a 15% to 20% range in the accuracies, as shown by the overlay, since it is a function of the actual maneuver being performed and the temperature variability at the gyro board. Increased performance can be achieved by shortening the interval between camera updates. At a 30 second update rate, the overall ISC accuracy (~50 arc-seconds in pitch/yaw, ~75 arc-seconds in roll) approaches the accuracy of the camera under typical conditions. Shorter camera update rates (< 60 seconds) increase the software throughput by 25%, the average operational power by 100 mW, and provide for a greater chance of optical interference.

ISC Design

The ISC consists of two separate units (as shown in Figure 3), connected by a cable: the Camera Gyro Assembly (CGA), which contains the sensors, and the Data Processing Assembly (DPA) containing the sensor's embedded computer and power supply electronics. The two-unit design facilitates integration with a host spacecraft. Only the CGA needs to be

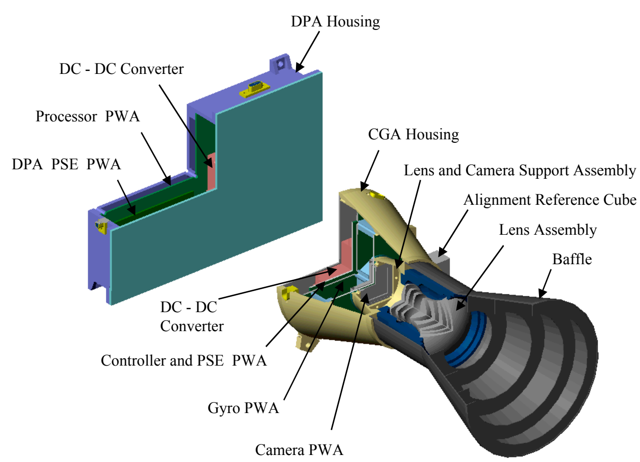


Figure 3: CGA and DPA Cutaway View

precisely aligned with the host spacecraft using the reference cube located on the CGA housing.

The CGA collects raw sensor data upon command and returns the data to the DPA for processing. The CGA and DPA communicate via a serial data link over an interface cable that can extend up to 2 meters. The CGA is 16 cm high (without mission specific baffle) and approximately 17 cm wide at its circular base. CGA mass is estimated at 1.2 kg. The DPA enclosure measures 15 cm x 23 cm x 4 cm with a mass of approximately 1.3 kg.

The CGA includes the lens, mission specific baffle, focal plane, three MEMS single-axis gyros with analog-to-digital conversion electronics, a controller board, and a 28V, triple output, DC/DC converter. The heart of the star camera is a STAR250 512x512 active pixel sensor array with an on-chip 10-bit A/D converter from Fill Factory. Advantages of the APS array over existing CCD technology include good radiation tolerance, no requirement for a thermal electric cooler, no requirement for a high voltage power supply, highly integrated digital control electronics, and low cost standard CMOS fabrication.[2,3] The camera has a command-ready interface to support windowing, various integration times, selectable frame count, and built-in test. The 21° square FOV star camera optic is a custom 35 mm, f1.2 lens manufactured by Zeiss and modified for space flight applications.

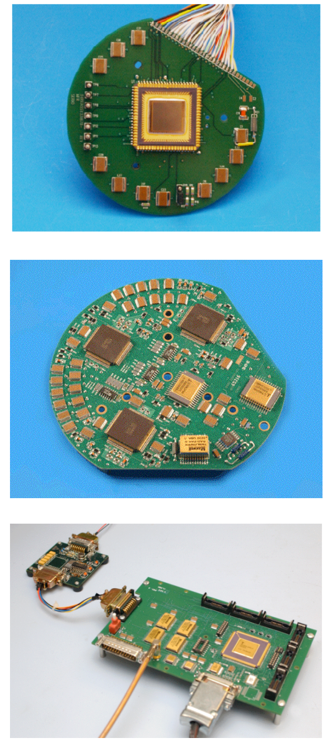


Figure 4: ISC focal plane (top); MEMS gyro board (middle); DPA electronics (bottom)

The CGA's controller board provides a simple serial interface to the DPA (or any other flight computer) and directs all necessary timing and control needed by the star camera and MEMS gyros. In addition, the controller provides 500 KBytes of SRAM storage to capture a full-frame compressed camera image for transfer to the data processor. Nominally, the image is transferred serially over the 16 MHz interface to the DPA allowing for full 512x512 pixel image transfer at 3.7 Hz. Given a faster interface to a controlling computer, the star camera and controller could support a full 30 frames per second, 512x512 pixel image transfer rate.

The Draper MEMS gyros sense angular rate by detecting the Coriolis effect on a sense mass. The tiny gyro sensors are etched in silicon using a Draper developed MEMS process. A sense mass is driven into oscillation by electrostatic motors. The mass oscillates in one axis and as the body is rotated, the Coriolis effect causes the sense mass to oscillate out of plane. This change is measured by capacitive plates and is proportional to the rotational rate of the body.

The DPA contains an Atmel ERC32 processor, power supply electronics, and a 28V, single output, DC/DC converter. The DPA interface to a host spacecraft is through a 3-wire, bidirectional, asynchronous RS422 serial port. Input rates are 9600 baud with a variable output data rate up to 38.4K baud. The large downlink capability of the ISC can support raw imagery from the star camera and also highly sampled raw and compensated gyro data transfer from the gyro electronics. The DPA processor can operate up to 32 MHz, but the ISC application clocks it down to 4 MHz to minimize power. Given a 4 MHz operating speed, the DPA can still provide sufficient throughput necessary to maintain accuracy given the infrequent camera updates and their associated image processing . Memory resources include 1 MB of EEPROM (to hold the operational flight code along with a master dark current frame, star catalog, and boot loader), 1.5 MB of EDAC SRAM (to execute operational flight code), and 6 MB of EDAC DRAM (for use as processor scratch pad memory, store up to 8 camera images, and 1096 sets of gyro data).

The ISC structure and its 50 Krad tolerant electronics are designed to accommodate launch loads and space radiation effects of Low Earth Orbit (LEO) and Geosynchronous Earth Orbit (GEO) spacecraft applications. Mechanical analysis was performed to General Environmental Verification Specification (GEVS) levels.

The cornerstone of the ISC design is its low power. As measured from hardware engineering models, CGA power is approximately 2 Watts (including DC/DC inefficiencies) with the camera continuously generating 512x512 pixel images at 3.7 Hz, three MEMS gyros reporting rate information at 320 Hz, and transmission of this data over an LVDS interface. The low power is possible due to the inherently low power requirements of the APS focal plane, MEMS gyros, and a prudent FPGA design. The space-ready DPA operating at 4 MIPS with floating point operations consumes less than 1.5 Watts of power (including DC/DC inefficiencies) due to the inherently low power of the ERC32 processor, 4 MHz operation, and a power saving FPGA design.‡

TECHNOLOGY VALIDATION

The overall goal for flight validation as stated by NMP is to reduce the risk and cost of using new technologies in NASA Space Science missions. Developing high fidelity models and validating them with ground and flight test data will accomplish this goal. The models allow the envelope of applicability to be extended to future applications that are not identical to the current experiment or technology.

Technology validation will be accomplished by a rigorous ground and flight test suite. Draper will conduct a series of analytical measurements, computer simulations, rate-table tests, star simulation tests, and observatory tests to validate the concept on the ground. These tests will characterize the ISC accuracy performance over varying camera update rates, angular slew rates, and temperature ranges with respect to power draw, computational throughput requirements, and memory storage requirements.

Table 1: Key Validation Objectives

Objective	Where	Metric	CBE
Accuracy (1-sigma) in each axis with slewing maneuvers	Ground & Flight	0.1 deg	<0.1 deg as function of camera update rate
Self-Initialize	Ground & Flight	<10 min over >90% of sky	<2 min over >99% of sky
Power	Ground & Flight	<= 4.5 W	~ 3.5 W
Mass	Ground	<= 3 kg	~ 2.5 kg
Space Qualified Component	Ground & Flight	Operates in specified environments	Designed for GEVS

‡ Further reading on the ISC design and operation is available to the reader.[4]

During flight, the ISC testing will validate predicted operation of the MEMS gyros (ARW, scale factor, and bias stability) in a space environment (micro-gravity, overall radiation exposure, micrometeoroids, and vacuum). Predicted imager performance (dim star limit, stray light rejection, attitude accuracy) will also be validated in the relevant space environment. The integrated performance of the MEMS gyros and star imager will be validated in space.

Validation Objectives

The key validation objectives are shown in Table 1. The CBE is given in the last column. Model accuracy and self-initialization CBE were verified through a series of analysis. The power and weight CBE are measured from actual hardware. The environmental testing to GEVS specification will start in 2003.[5]

Ground Validation

Ground validation is divided into six major areas: gyro performance, camera performance, integrated simulation, software tests, observatory tests and environmental tests.

Gyro performance tests will verify that the scale factor error, angle random walk, turn-on bias, time and temperature bias stability, axis alignment and stability, gyro analog/digital noise and quantization can meet the gyro subsystem error requirements of less than 300 arc-seconds.

The camera performance tests will verify that the detector noise, interpolation error, image smearing, lens-detector misalignment, line scan aberration, number of stars and their distribution in the FOV, and other minor error sources (including chromatic aberration and calibration residuals) can meet the camera subsystem error requirements of less than 75 arc-seconds.

The integrated simulation includes a computer simulation of the camera, gyro, and Kalman filter. This will verify that the gyro and camera performance measurements combined in the Kalman filter can produce a single ECI quaternion of 0.1° accuracy under nominal and worst case conditions.

In the engineering model test environment, "canned" images will be collected using the actual ISC camera and electronics. Based on the gyro performance measurements, a gyro simulator will generate high fidelity, repeatable, motion profiles to verify system accuracy under most any conditions as virtually any input parameter can be varied to determine the system response. This environment will also be used to test off-nominal and fault conditions.

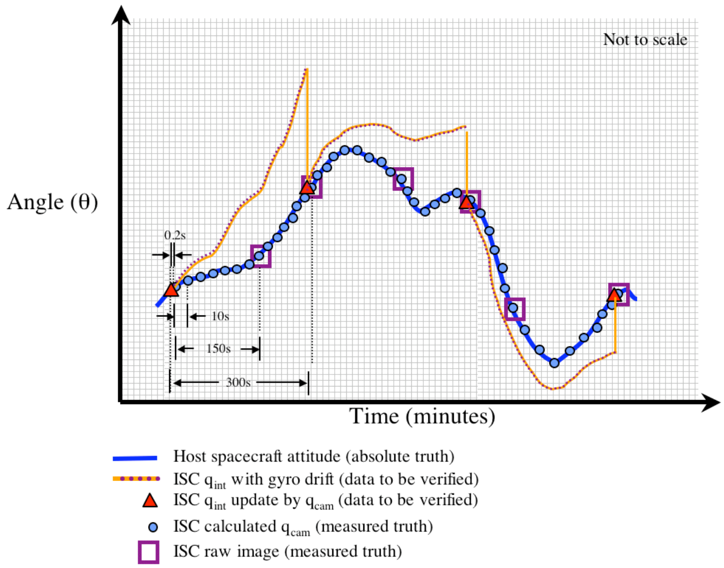
Observatory tests will verify overall ISC system accuracy and self-initialization. The flight ISC will be mounted on a 24-inch telescope at the Wallace Observatory in Westford MA. The telescope (and ISC) will be slewed and then positioned for camera updates. The telescope has 2 arc-second knowledge and an ability to slew up to 1.5°/second. The RA/DEC output of the observatory will be recorded, converted to an ECI quaternion, and compared with the ISC quaternion to

Figure 5: MIT's Wallace Observatory located in Westford, MA

determine the overall system accuracy. Mission test cases very similar to those that will be conducted during flight validation with the host vehicle (but limited to the Northern hemisphere) will be established and run at the observatory.

Finally, the flight ISC will undergo an extensive environmental test program including GEVS level random and acoustic vibration, shock, thermal vacuum, and MIL-STD-461 EMI/EMC.

Figure 6: ISC Self-Score Concept



Space Validation

The ISC's performance will be verified in space over the same varying camera update rates and angular slew rates that were conducted during ground tests. The performance in space will be correlated to the accuracy and power measurements taken on the ground. The ISC power consumption and self-initialization time will be monitored over various sky quadrants and angular rates.

For accuracy validation, two of the three sources of truth are produced by the ISC itself. This concept is referred to as self-score. The ISC attitude quaternion to be validated, shown by the yellow dots in Figure 6, is output at 5 Hz and sent to the ground. In the self-score concept, there are two sources of truth attitude for comparison to the ISC quaternion; ground processed raw images and the ISC camera quaternion.

The first source of truth is the raw images captured, downlinked, and processed on the ground. These are shown by the purple squares in Figure 6. The LIS algorithms will independently calculate a camera quaternion from the raw image for a comparison to the ISC quaternion at that point in time.

The second source of truth is the ISC camera quaternion. This is the on-board ISC processed camera data without any gyro data (as shown by the green circles in Figure 6). For validation, the camera will be commanded to acquire an image at short intervals and process the LIS algorithm in the ISC. These short interval camera quaternions will not update the ISC quaternion, except at the camera update interval, and will be forwarded to the ground.

The third source of truth is the host spacecraft attitude for which a time-tagged independent attitude representation will be generated and sent in telemetry to the ground. This data will be correlated to the data generated onboard the host spacecraft.

Draper will develop a report summarizing the ground and flight validation results as required by the NMP for ST6. In the report, numerous details of the mission will be documented along with an overall assessment of the ISC's technical performance.

NEXT GENERATION ISC PERFORMANCE ENHANCEMENTS

Several enhancements have already been identified that will substantially improve the performance and functionality of the ISC. These represent the next generation of integrated attitude and navigation hardware. The benefits and costs of each will be examined to assess which are worth developing and incorporating into the next generation ISC sensor.

Improving MEMS gyro performance will improve the system performance. The development of the Draper MEMS gyros is ongoing and performance continues to improve as the design evolves. Draper is a leader in MEMS gyro development and is constantly improving both the sensor and electronics design. Next generation MEMS gyros currently in development will be fabricated with an upper sense plate (USP) to reduce noise.[6] This improvement is expected to reduce the angle random walk to half of its current value resulting in significant system level improvements in performance.

APS detector designs continue to evolve as well with improvements such as larger arrays, better process uniformity, and improved quantum efficiency. Larger arrays such as the Fill Factory 1024x1024 pixel array are already available.[7] Larger arrays can improve the resolution of the detector. As CMOS processes evolve, feature uniformity improves and feature size decreases. As feature size decreases, quantum efficiency is improved as features occlude less of the APS pixel well.

Another worthwhile design improvement is to provide the ability to interface two CGAs to a single DPA. This feature was not part of the baseline design, but would be of value to space missions, especially when higher performance is required. On average, star tracker boresight performance is worse than the performance across the field of view. For this reason, two trackers are used for most missions. Allowing two CGAs to interface to a single DPA would eliminate a DPA and reduce mass and power requirements further.

Improving the mechanical alignment stability of the ISC to the host spacecraft will also improve performance. For example, a kinematic mount could be used to obtain better mounting accuracy and reduce the amount of alignment error between the ISC and the host spacecraft.

Novel electronics packaging techniques are being explored to further reduce size and add functionality to the ISC. Draper recognizes that as the feature size of these devices continues to shrink, new and different ways of packaging are required. Ball Grid Array, chip-scale packaging, Chip on Board, and Flip Chip methods are ways that would provide more efficient interconnection and therefore increase the overall ISC packaging efficiency.[8] Envisioned is an integrated DPA module that combines the ERC32 microprocessor, memory, and glue logic in a stacked module that is estimated to be 2 inches on a side.

Incorporating MEMS accelerometers would provide a full MEMS-based Inertial Measurement Unit (IMU) and introduce a basic navigation capability to the ISC. Draper, as well as being a leader in MEMS gyro development, has also developed high performance MEMS accelerometers that have been proven in extensive and severe environmental testing. These accelerometers could be integrated alongside the gyros in the CGA. Beyond the addition of a miniature MEMS accelerometer triad to the ISC, one could also add a miniature GPS receiver to ISC to provide a full navigational capability in LEO mission orbits. These additional hardware elements could be integrated into the ISC with minimal size, power, and mass increases through a combination of Ultra-Low Power (ULP) electronics along with advanced packaging/interconnect techniques.

Algorithms that identify and track the stars are constantly improving and could result in faster acquisition and initialization times. The specific Mortari algorithm and its implementation in the ISC will be tuned based on test results of the validation system for better performance and to minimize the associated resources.

TECHNOLOGY INFUSION

Technology infusion is the process of providing a new and useful technology to NASA missions that benefit from the innovation. The ST6 ISC flight experiment not only provides the mechanism to verify and benchmark pre-launch performance predictions, but also serves as a showcase for technology possibilities early enough in the ISC product development cycle to easily adapt to changing requirements.

The foundation of the infusion process is fostering an understanding and interest in the ISC's capabilities. This motivates the need to initiate and maintain a continuing contact with the potential mission customers and stakeholders to assure a clear understanding of technology needs and infusion opportunities. This involves seeking out candidate missions and fine-tuning the product to be as widely beneficial as practical. It is expected that the ISC will offer designers an attractive alternative for challenging applications where power, mass, and volume constraints preclude the use of the traditional star tracker and inertial sensing components. It is envisioned that many of NASA's future missions, ranging from satellites in LEO to deep space interplanetary probes, could benefit from the infusion of the versatile ISC technology in some manner.

The infusion process begins with a survey of missions currently in formulation and longer-term "Roadmap" missions to establish both the attitude determination requirements and the hardware resource constraints. A critical element of this survey process will be technical interaction with NASA Code S (Space Science) Theme Technologists to understand their theme-unique technology needs. The NASA Earth Science Technology Office (ESTO) would also be solicited to understand the Code Y (Earth Science) platform technology needs for attitude determination. Also, the Integrated Mission Design Centers at both GSFC and JPL, where collaborative end-to-end mission conceptual design studies are performed, will also serve as rich sources of requirements and constraints data.[9] An analysis of this information should identify specific missions where the ISC technology is most applicable, providing the biggest "payoff".

When the ISC technology appears to be applicable to a mission, direct contact will be made with that mission's Program Office, formulation team, and/or lead GN&C engineer to explore the opportunity for potential infusion. In some cases the ISC technology may be a direct "as is" fit, whereas in other cases it may be close to meeting mission requirements.

As described above, several useful design enhancements have been identified that will substantially improve the performance and functionality of the ISC. For those missions where the ISC is close to meeting the requirements, this set of potential design enhancements will be explored. Specific enhancements to the ST6 ISC baseline design will be studied and evaluated to determine the cost versus benefit trade space. Key trade parameters would focus on its size, mass, and power and would drive the decision on which modifications are worth pursuing for a specific mission application.

In addition to efforts to seek out missions that would benefit from ISC, information about ISC will be published. The NMP website will be updated with the most up to date information regarding the ISC capabilities and development status. Contact information will be provided for those interested in learning more about the vast potential of the ISC. Also, the ISC will be added to the technology item database in the Integrated Mission Design Centers at both GSFC and JPL so that this innovative technology is available to the spacecraft designers conducting end-to-end mission conceptual design studies.

Certainly, following successful ST6 on-orbit validation and subsequent commercialization of the technology into an affordable product form, one can readily foresee a path for rapid adoption of the ISC by NASA, DoD, and Industry as a uniquely enabling technology to perform the micro-satellite attitude determination function. While that particular ISC use will address a very critical and specific need, it is difficult to imagine the infusion of this technology being limited to that single micro-satellite application. While it is unreasonable to predict that the ISC will be a panacea solution suitable for all next generation missions, there is clearly a wide-ranging set of future GN&C problems for which the ISC technology is relevant and may represent part or all of the solution. It is entirely possible that the ISC technology, in either its initial ST6 configuration or in a slightly modified form, will be employed to address some of the GN&C design challenges posed by: highly dynamic vehicles, tumbling and/or "lost in space" spacecraft, semi-autonomous (tele-operated) satellite servicing platforms, autonomous rendezvous vehicles, reusable launch vehicles, formation flying platforms, deep space probes, and planetary exploration vehicles (surface rovers, sample return craft, aerial platforms, and balloons).

MISSION APPLICATIONS

The success of NASA's next generation Earth and Space Science missions will be critically dependent on the development, validation, and infusion of spacecraft avionics that are not only highly integrated, power efficient, and minimally packaged, but also flexible and versatile enough to satisfy multi-mission requirements. The imminent introduction of the ISC into the spacecraft designer's inventory will directly address this need for advanced technology avionics and will herald a breakthrough in how the function of medium (arc-minute class) accuracy attitude determination will be implemented in future space missions. It is perhaps the inherent multifunction capability of the ISC that is of greatest relevance to future space missions. Technologists at Draper and NASA understood the powerful advantages to be gained in power, mass, and volume by combining the traditionally separate gyro (inertial) sensing and stellar sensing functions into one unique ISC package. The APS star sensor and the MEMS gyro technology combination described above provides the innovative technology "push" for ISC. Conversely, the primary customer "pull" for ISC is derived from NASA's need to implement affordable, next generation, micro-spacecraft capabilities. In short, the ISC will not only support the on-going general evolution towards smaller, lower power spacecraft sensors, but also will directly focus on and address the technology needs for the on-going micro-satellite revolution.

It is envisioned that the ISC will serve as both an *Enhancing* and an *Enabling* technology for many future Space Science and Earth Science missions. Enabling technologies are those that provide the presently unavailable capabilities necessary for a mission's implementation and are vital to both NASA's intermediate and long term missions. Enhancing technologies typically provide significant mission performance improvements, mitigations of critical mission risks, and/or significant increases in mission critical resources (e.g., cost, power, and mass).

A number of potential general mission applications for the ISC technology are described below. Mission applications directly addressed by the baseline ST6 ISC configuration will be described, then ambitious mission applications, where hardware and software modifications to the baseline ISC would be required, are subsequently discussed.

The Microsat Application

One of NASA's top technology priorities focuses on economical, highly autonomous, micro-satellites for multi-spacecraft distributed missions. The ISC represents an enabling, fully integrated, micro-satellite attitude determination system that directly addresses this NASA priority. Preliminary analysis of a three-axis stabilized 50kg micro-satellite conceptual design indicates the critical need for the ISC technology to meet stringent power, mass, and volume requirements while simultaneously satisfying a 0.03-0.05 degree attitude determination requirement. In this conceptual design, the ISC is a fully compatible external sensor interfaced to the Multi-Function GN&C System (MFGS) currently being developed at NASA/GSFC.[10] The versatile MFGS unit represents the successful coalescence of several hardware and software technology innovations into one single, highly integrated, compact, low power, and low cost unit that simultaneously provides attitude determination solutions and navigation solutions with accuracies that satisfy many future GSFC micro-satellite mission requirements.

The Satellite "Single Sensor" Application

Mission designers could employ the ST6 configuration ISC unit as a significant enhancing technology for a number of moderate (< 0.1°) non-microsatellite applications. In these cases, the ISC would serve as a compact and low power "Single Sensor" unit that could replace typical attitude sensors and associated attitude determination flight software.

For example, one could foresee the use of the ISC on some GEO communications or observation satellites replacing the typical attitude sensor complement of horizon crossing indicators, Earth sensors, and Sun sensors. Mass savings in this particular geosynchronous application would directly translate to additional station-keeping propellant or payload capacity.

The ISC seems uniquely suited to fill a medium (arc-minute class) accuracy attitude sensor technology niche between relatively low cost/low performance COTS Sun Sensors/Magnetometers and relatively high cost/high power COTS Star Trackers with arc-second performance.

The Satellite Safe-Hold/Coarse Acquisition Application

Identifying and implementing simple, reliable, independent, and affordable (in terms of cost, mass, and power) methods for autonomous satellite safing and protection has long been a significant challenge for designers. Employing one or as many as three (if an attitude voting scheme is desired) "bolt on" ISC's to provide a fully independent safe-hold attitude determination system for high end (very costly) one-of-a-kind observatories, such as the James Webb Space Telescope (JWST), may yield a simple solution to a complex problem. Specific implementations would vary, but in general, this application entails one or more of the ISC units being mounted on the observatory to mitigate the risk of mission loss for a small relative cost. ISC represents an enhancing technology in this application. The low mass and small volume of the ISC should preclude any major accommodation issues on a large observatory. Additionally, the performance of the baseline ST6 ISC configuration can easily satisfy the typically modest safe-hold attitude determination requirements (on the order of a degree or less) to maintain the observatory in a power-safe and thermal-safe attitude.

One notional safe-hold operational concept calls for the ISC units to be powered off at launch. They would then be autonomously powered-on upon indication of observatory separation from its launch vehicle. In this state, the outputs of the ISC's MEMS gyros alone could provide sufficient angular rate polarity feedback to stabilize the observatory if in a tumbling state due to high tip-off rates at separation. The ISC can also provide a coarse attitude acquisition function prior to handing over attitude determination to the observatory's precision attitude sensors or, in some cases, the payload's Fine Guidance Sensor (FGS). The ISC units would subsequently be powered-off once the nominal observatory attitude state is achieved. They would be left in an unpowered state during normal mission operations and would only be activated when a safe-hold event is triggered by the observatory's FDC logic. The low power draw of the ISC would be very compatible with the initial phase of safe-hold operations when only very limited battery power may be available. The outputs of the MEMS gyros alone could provide sufficient angular rate polarity signal feedback to stabilize the observatory if in a tumbling state following an on-orbit anomaly. Once the anomaly is resolved, the ISC could perform the coarse attitude determination function supporting the transition back to normal mission operations.

MODIFIED ISC CONFIGURATION MISSION APPLICATIONS

In addition to the mission applications discussed above, there are a number of other potential uses of the ISC technology that can be conceived. The Draper/NASA team has conceptually identified, but not studied in any significant detail, the following potential next-generation ISC technology infusion targets.

The GEC Mission Application

Planned for launch in 2009, the Geospace Electrodynamics Connections (GEC) mission will be an element of the NASA's Sun-Earth Connection (SEC) Space Science theme.[11] GEC will make in-situ measurements in the Ionosphere-Thermosphere region to improve models of how the Earth's upper atmosphere electrodynamically couples to the magnetosphere. In this discussion, it is important to note that the GEC spacecraft with an on-orbit mass of approximately 375 kg is not a micro-satellite class vehicle. In fact, a considerable amount of propellant will be carried on GEC to perform "deep-dipping" campaigns down to perigee orbital altitudes, as low as 130 km, where the mission's primary science data is taken. The amount of science data collected overall is a function of the number of dipping campaigns, which itself is a function of the amount of propellant. The current baseline GEC Attitude Determination and Control Subsystem (ADCS) design incorporates a COTS star tracker unit, an Inertial Reference Unit (IRU), sun sensors, and customized Kalman filter flight software algorithms hosted on the spacecraft's central processor to satisfy the attitude determination requirement of 0.01 degree. The performance of the baseline ST6 ISC technology is not currently predicted to satisfy the GEC attitude knowledge requirement of 0.01 degree but that level of required performance could be achieved with technology enhancements. The ISC could, with a minimum of modifications, serve as the "Single Sensor" replacing the individual COTS Star Tracker and the IRU.

It is estimated that approximately 1.5 kg of mass and 26 Watts of power can be saved by employing a single ISC unit rather than the separate COTS Star Tracker and IRU. The mass savings afforded by using the ISC to perform the attitude determination function could be allocated for additional propellant, thus more "deep-dipping", and thus more GEC science data. That is one advantage of using ISC on GEC. The other has to do with power and it looks to be even more significant than the mass advantage.

The design of the GEC Electrical Power Subsystem (EPS) poses significant challenges. The ADCS is the largest power consumer of all the spacecraft bus subsystems. Controlling the spacecraft while "deep-dipping" at perigee may demand as much as 100 Watts of power for the ADCS. This is about twice the ADCS power requirement while operating in the non-perigee portions of the mission orbit. The large perigee ADCS power draw is primarily driven by the need for the reaction wheels to generate attitude control torques to counteract the significant aerodynamic disturbances acting on the spacecraft during perigee passage. The complicating factor for the GEC EPS designers is the added difficulty in generating electric power during perigee passage. Using the ISC significantly lowers the ADCS power requirement (by approximately 26 Watts) during perigee passage thus reducing the demand on the EPS and/or making more power available for the GEC science payload.

Lastly, there is another, more subtle, but significant, advantage of using the ISC technology on a mission such as GEC. As described in the System Overview above, the ISC contains all the necessary flight software algorithms embedded in its dedicated internal processor. The fact that customized Kalman filter flight software attitude determination algorithms would not have to be designed, developed, tested, and integrated into the spacecraft central processor should serve to reduce mission costs and risk.

Although several implementation details remain to be studied, a strong compatibility of the ISC technology with the GEC mission requirements has emerged from the preliminary top-level considerations. The GEC represents a very solid potential technology infusion target for ISC that merits additional consideration.

The Autonomous Visual Relative Navigation Sensor Application

In order to accomplish its ambitious mission goals for the next two decades there is a heightened priority within NASA to develop and infuse the multiple technologies needed to autonomously perform both the functions of: 1) multiple, distributed spacecraft formation flying and 2) spacecraft rendezvous, proximity operations, and docking. Several missions currently being formulated at NASA rely on the capability to fly multiple distributed spacecraft to enable revolutionary methods of Space and Earth Science observation.[12] Likewise, NASA foresees several important applications of the capabilities for spacecraft rendezvous, proximity operations, and docking such as on-orbit deployment, inspection, assembly, servicing, re-fueling, payload retrieval/transfer (e.g., sample canisters and/or experiment packages), and re-configuration at various orbital locations.

The ultimate success of the planned formation flying and rendezvous missions will depend on the development of advanced electro-optical sensors (and algorithms) for performing vision-based relative navigation. The ISC technology is a natural candidate for this type of application and could be readily adapted with a minimum of camera and flight software modifications.

The ISC could be a coarse acquisition visual sensor to determine the relative location of multiple formation spacecraft immediately following deployment from the launch vehicle dispenser. Although there are a number of passive and active techniques that could be used in this rendezvous application, it was assumed in this concept that each individual spacecraft is equipped with a uniquely identifiable array of small, perhaps actively modulated, Light Emitting Diode (LED) beacons that can be sensed by the ISC detector. In this specific application, the ISC would first be used to provide coarse navigation outputs of sufficient accuracy to simply ensure collision avoidance between the spacecraft. Simple relative navigation algorithms would need to be embedded on the ISC processor.

In a similar manner, the ISC technology lends itself to the rendezvous problem of first acquiring and subsequently sensing the relative position and attitude of the orbiting target spacecraft by the maneuvering chase spacecraft. In this application, vision-based feedback from the ISC is used in an integrated orbital/attitude control loop on the maneuvering chase spacecraft. It is conceivable that the ISC could potentially serve as the relative navigation sensor for all phases of the rendezvous and docking process: long range acquisition and tracking as well as close range relative orientation and position determination.

Long range initial acquisition of the target vehicle could be performed passively by the ISC by detecting the target as a moving point source against the inertially fixed star background. The appropriate relative navigation filter algorithms would need to be hosted on the ISC internal processor to accomplish this task.

Close range maneuvering of the chase spacecraft towards terminal engagement with the target could then be performed using the attitude and position derived from object feature (model matching) recognition processing algorithms using a series of ISC-generated video image inputs. Potentially these image processing algorithms could be hosted on the internal ISC processor, but that would be dependent on tailoring their size and complexity.

Once close enough to conduct proximity operations, the multipurpose ISC can serve as a high bandwidth video camera system to conduct detailed inspections of the target spacecraft prior to performing any docking, servicing, re-fueling, or payload retrieval operations. Conceivably two ISC's could be employed to generate stereo images of the target vehicle.

The Nanosatellite Core Bus Application

A novel and promising application is to use the lightweight, highly compact, and low power ISC as the enabling technology core of a 10-15 kg nano-satellite class bus architecture. It would inherently satisfy the attitude determination requirement and the remainder of the vehicle could be packaged around it by employing other advanced technologies for miniature, highly integrated, power efficient electronics. MEMS technology could be exploited in this architecture to implement many of the subsystem functions creating flexible and versatile design for multi-mission applications.[13] The ISC camera system could conceivably serve as the imaging payload in a two CGA/one DPA hardware configuration. With some hardware enhancements to the ISC's internal processor, it could serve as the nano-satellite central flight processor taking on the additional computational tasks required to perform the payload, command/telemetry, communications, power management, I/O, and attitude control functions. Additional interfaces would be necessary for subsystem control elements such as nano-wheel or MEMS thruster attitude control actuators.

CONCLUSION

An attitude determination sensor has been introduced and shown to be useful to a broad range of spacecraft as either a primary attitude sensor or as a safe-hold sensor. For moderate performance micro-satellites, ISC provides an enabling technology. The ISC promises benefits of ultra-low power and low mass by integrating state of the art complimentary sensors and an embedded processor in a compact, efficient package. An additional benefit of ISC is simplified spacecraft integration.

As a "bolt-on" smart sensor, the ISC reduces complexity by tightly integrating the gyros and APS star camera within the CGA to simplify alignment and calibration. Integration costs are reduced since the DPA performs all of the attitude determination and filtering with a simple serial interface to provide the spacecraft with attitude information in the form of an attitude quaternion and attitude rates. To further reduce size and power impact on the spacecraft, the ISC design also offers the possibility of using its data processor as the spacecraft's primary processor to run both the ISC and spacecraft software.

Preliminary analysis indicates that a multitude of spacecraft will benefit from the ISC technology infusion process, pending specific design trades. One objective of this paper was to stimulate ideas for infusion and adoption of the ISC technology in the minds of future mission designers. It is exciting to contemplate the various mission applications this breakthrough ISC technology might support in the near to mid-range future.

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ACRONYMS

ADS	Attitude Determination System
APS	Active Pixel Sensor
ARW	Angle Random Walk
ASIC	Application Specific Integrated Circuit
CBE	Current Best Estimate
CCD	Charge Coupled Device
CGA	Camera Gyro Assembly
CMOS	Complementary Metal Oxide Semiconductor
DPA	Data Processing Assembly
ECI	Earth Centered Inertial
ISC	Inertial Stellar Compass
FDC	Fault Detection and Correction
FOV	Field of View
FPGA	Field Programmable Gate Array
GEC	Geospace Electrodynamics Connection
GN&C	Guidance, Navigation, and Control
GSFC	Goddard Space Flight Center
LIS	Lost-in-Space
LVDS	Low Voltage Differential Signal
MIPS	Million Instructions Per Second
NASA	National Aeronautics and Space Administration
PPM	Parts Per Million
NMP	New Millenium Program
GEO	Geosynchronous Earth Orbit
GEVS	General Environmental Verification Specification
GPS	Global Positioning System
LEO	Low Earth Orbit
MEMS	Micro-ElectroMechanical System
MFGS	Multifunction GN&C System
RA/DEC	Right Ascension and Declination