A MICROMECHANICAL GYRO PACKAGE WITH GPS UNDER DEVELOPMENT FOR SMALL POINTING SATELLITES

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ABSTRACT

The cost and complexity of large satellite space missions continue to escalate. To reduce costs, more attention is being directed toward small, lightweight satellites where future demand is expected to grow dramatically. Draper Laboratory is currently developing a three-axis stabilized control system using on-board GPS, micromechanical gyroscopes, small reaction wheels, and miniaturized microstrip phased array antennas for active line-of-sight control of a small pointing satellite. A first iteration of the rate gyroscope electronics has been designed and implemented on a single mixed-signal CMOS ASIC. The gyroscope sensor, ASIC, and supporting components are placed in a 2.5-cm X 2.5-cm flat package. Total power dissipation for the gyroscope ASIC is a small fraction of a watt. The micromechanical gyro data is optimally blended with the GPS data, thereby taking advantage of the gyro's ability to measure high-frequency dynamics and the GPS's ability to bound the error growth due to gyro drift. First-order simulation has shown that the performance of the micromechanical gyros, when integrated with GPS, is feasible for a pointing mission of 1 microradian of jitter stability and ~2 milliradians (1σ) absolute error, for a satellite with 1 meter antenna separation.

INTRODUCTION

The cost and complexity of conducting space missions with individual, highly integrated, large satellites continue to escalate. To reduce costs, more effort and attention is being directed toward small, light-weight satellites where future demand is expected to grow dramatically. During the past few years significant effort has been made to develop the "smaller, better, faster" approach being embraced by the aerospace industry as a means to prepare for space missions of the future. Existing small satellite pointing systems do not employ active control to improve pointing performance. Some systems allow the bus to point coarsely, then attempt to maintain the image on the focal plane with fast-steering mirrors. Image smear can be reduced by on-board attitude determination from inertial sensing, coupled with GPS or GLONASS, and closed-loop error correction and attitude control actuators. However, traditional inertial reference systems could consume a significant fraction of the mass budget for a micro-class satellite. Recent advances in silicon microfabrication technology have led to the development of low cost micromechanical inertial sensors. The small size, low weight, and low cost attributes of these sensors permit on-board insertion of gyroscopes and accelerometers for space applications previously considered impractical.

Draper Laboratory is currently developing a three-axis stabilized control system using on-board GPS, micromechanical gyroscopes, small reaction wheels, and miniaturized microstrip phased array antennas for active line-of-sight control of a small pointing satellite.

The feasibility of the micromechanical gyro to provide sufficient performance to accomplish a small satellite pointing mission has been analyzed for an attitude control bandwidths of 0.15 and 0.5 Hz, based on estimated external and induced disturbance frequencies. From a first-order simulation, it was determined that the performance of existing micromechanical gyros with integrated GPS is capable of providing pointing jitter stability of approximately 1 microradian and absolute pointing error of approximately 2 milliradians (1σ), for a satellite with 1 meter antenna separation (i.e., 1m baseline). The absolute pointing error is a factor of three better than attitude control by GPS alone. Similar results would be expected if GLONASS were used in place of, or in combination with, GPS. In the remainder of the discussion that follows, it is assumed that GPS or GLONASS could be used interchangeably.

POINTING MANEUVER

A typical pointing maneuver requires the satellite to process GPS-derived attitude measurements (and perhaps attitude-rate) and gyro-derived attitude rate measurements while in orbit to establish estimates of gyro drift bias and scale factor (SF). Satellite maneuvers can be used to separate the error components due to bias and SF.

The satellite is pitched forward from a nominal nadir-pointing attitude to some new attitude, and then pitched in reverse during imaging to reduce the transverse velocity of the line-of-sight at the surface of the earth (see Figure 1). Without this counter motion, the satellite velocity over an integration period will result in too much blur of the picture. The amount by which the satellite must be initially pitched forward in preparation for the imaging slew depends upon the amount of time required to settle out initial pitch slew transients once the imaging slew begins. The time requirement, in turn, depends upon the controller bandwidth. A higher controller bandwidth will result in transients subsiding quicker but will result in a greater jitter response to sensor noise. A 0.15 Hz controller bandwidth requires an initial attitude θ of a least 60° off the nadir, and a 0.5 Hz controller bandwidth requires less than 30°, in order to satisfy the requirement that the additional drift (i.e., in addition to the nominal drift due to satellite motion) over an integration period should not exceed the equivalent of 1 pixel motion at the ground.

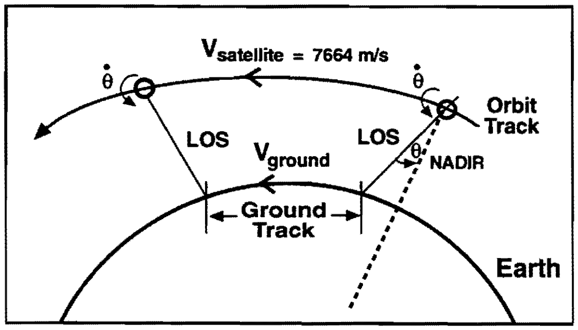

Figure 1. Imaging Mission

During this large angular rotation in a short period of time there will likely be several changes of GPS satellites and, possibly, a large variation of attitude precision. Therefore, attitude measurements from GPS will be effectively suppressed during the imaging slew, and attitude inputs to the controller will rely essentially upon calibrated gyro measurements. Calibration of the gyros depends upon the effectiveness of the integrated GPS/gyro filter. If gyro calibration is not sufficiently accurate, pausing at the end of the preparatory pitch-forward maneuver will regain the required absolute attitude accuracy by again processing GPS attitude measurements at the new pitch attitude.

3-AXIS MICROMECHANICAL GYRO PACKAGE

Draper Laboratory has been developing micromachined silicon gyroscopes and accelerometers since 1984. Figure 2 illustrates the steady improvement in gyro resolution performance since 1990.

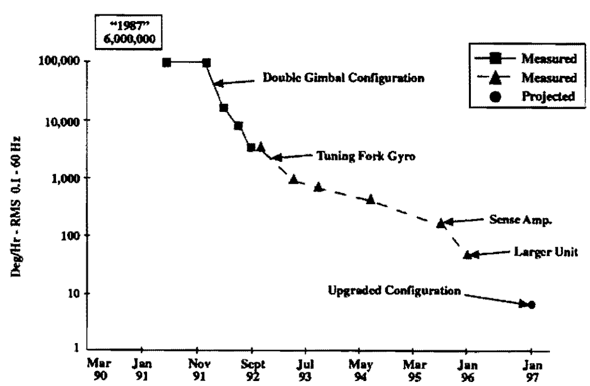

Figure 2. Micromechanical Gyro Performance History

The micromechanical gyros (Figure 3) are of extremely small size, with each sensor occupying an area of less than 4 square millimeters. These sensors are extremely rugged, and easy to fabricate. They are microfabricated from single-crystal silicon anodically bonded to a glass substrate (dissolved wafer process). The gyro senses the Coriolis acceleration of the proof masses operating as a double-ended tuning fork. Sense and drive are electrostatic. These instruments are described in more detail in the bibliography.

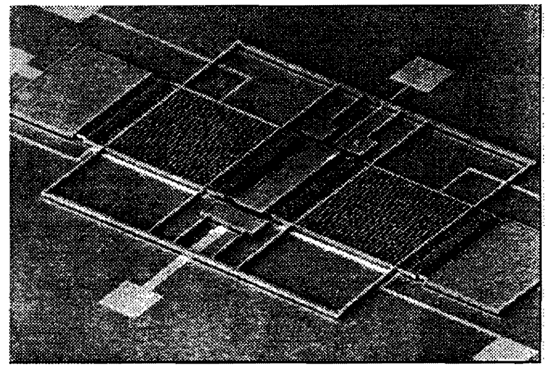

Figure 3. Micromechanical Tuning Fork Gyro

A miniature INS/GPS system is being developed at Draper for terrestrial guidance applications. The system contains a tactical-grade Micromechanical Inertial Measurement Unit (MMIMU) comprising three orthogonal micro-mechanical accelerometers and three orthogonal micromechanical gyros with individual conditioning electronics and high-resolution digital output quantizers. The system also contains a GPS receiver as well as a guidance and navigation processor. This entire system is miniaturized further using low power mixed signal CMOS ASICs, high-performance bump bonding techniques, shared/multiplexed electronics functions and advanced high-density packaging. A single clock, voltage reference, A/D converter, and DSP are used to service the micromechanical INS/GPS system, resulting in a compact, low-power, multi-sensor, multi-axis system. Power dissipation is minimized both through the use of low power CMOS and also by using power conserving sleep modes. Miniaturized GPS microstrip antennas, fabricated on high-dielectric constant ceramic substrates, permit significant reduction in size compared to antennas fabricated on common low-dielectric substrates. The 3-axis μgyro package for space applications is a more accurate derivative from this MMIMU development activity.

The preliminary 3-axis μgyro package comprises three orthogonally mounted 2.5cm x 2.5cm x 0.5cm flatpacks (Figure 4), each containing one mixed-signal CMOS ASIC, power conditioning and trim circuitry, and a single degree-of-freedom TFG. Total power dissipation of the flatpack is a small fraction of a watt. Present units have demonstrated 0.5°/s bias stability over a temperature range of -40 to +85°C without thermal compensation or control, and bias stability with temperature control of 1°/hr and angle random walk <0.1°/√h. This preliminary package is being donated to the JAWSAT experiment.

The future evolution of these sensors is the fabrication of three orthogonal gyros on one substrate. Multi-chip modules (MCMs) and mixed-signal application-specific integrated circuits (ASICs) are being developed further for the implementation of micromechanical (MEMs) systems. Flip-chip (or bump) bonding is required for alignment accuracy between sensors and to fiducials (reference lines and edges or surfaces of circuit board upon which components are mounted).

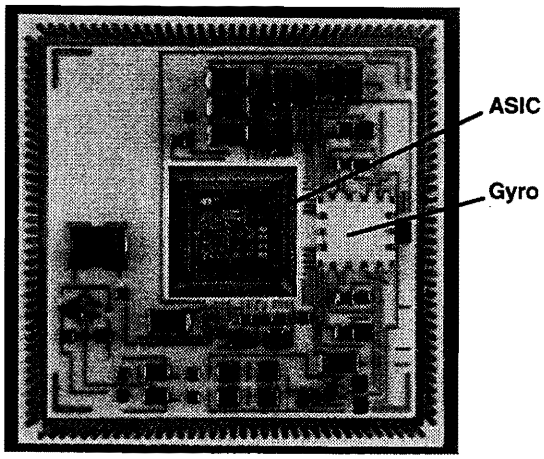

Figure 4. 2.5-cm by 2.5-cm Gyro and ASIC Package

3-AXIS µGYRO PACKAGE/GPS INTEGRATION

Figure 5 describes the pointing satellite µgyro package/GPS integration in which the flight processor blends the µgyro/GPS data to determine translational and rotational state.

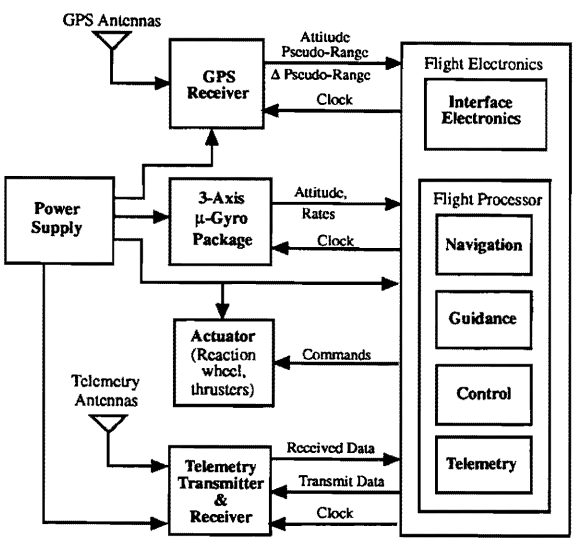

Figure 5. 3-Axis µGyro Package/GPS Integration

GPS Role

GPS provides position with accuracy on the order of a few meters, and provides a highly reliable technique for providing attitude using interferometry techniques. GPS has the ability to measure velocity changes very precisely via carrier tracking, and is therefore used in place of accelerometers; however, antenna masking and satellite observability concerns still remain.

The determination of attitude using GPS is based upon the carrier phase difference between two antennas tracking the same satellite. Figure 6 illustrates the basic measurement geometry. The angle θ is determined by knowing the baseline (**b**) in body axis and measuring the GPS signal delay ($\Delta\mathbf{r}$). Thus, θ can be determined as follows:

$$\theta = \cos^{-1}(\Delta\mathbf{r}/\mathbf{b})$$

Extending this to three axes using more than one baseline allows complete determination of vehicle attitude

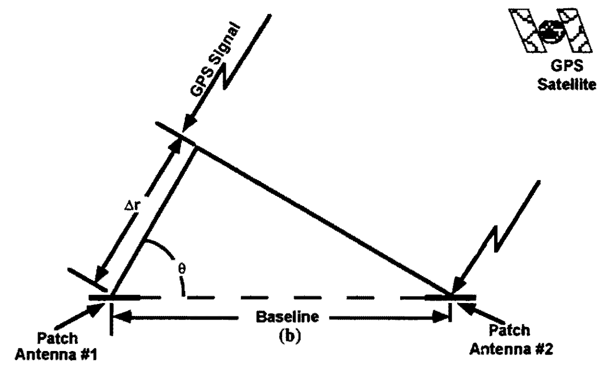

Figure 6. GPS Interferometry

One example of this technique is the GPS Attitude and Navigation Experiment (GANE), which tested a GPS interferometer intended for International Space Station Alpha (ISSA). The experiment flew onboard the Shuttle in May 1996. A four-antenna interferometer was mounted on a 1.5m by 3m platform in the cargo bay, along with an Inertial Reference Unit (IRU) for attitude verification. The requirements for this stand-alone GPS interferometer were to estimate the station's attitude to within 0.3 deg (3σ) per axis and attitude rates to within 0.01 deg/s (3σ) per axis at 0.5 Hz. Preliminary results indicate that the experiment was successful.

The ability to determine the translational and rotation state of a satellite is dependent on the number of GPS satellites in view and the resulting measurement geometry. Since the GPS system was designed for "Earth-bound" users, the GPS satellite radiation signal is directed toward Earth. Satellites at altitudes up to approximately 1000 km will generally find 10 or more satellites "in view". At higher altitude, the user may be outside the main radiation beam of some GPS satellites. At altitudes above roughly 3000 km, the number "in view" may frequently drop below 4. Although 4 satellites in view is generally considered the minimum set to solve for both the three components of position and the user clock bias, studies have shown that acquisition of a single GPS satellite can be adequate to maintain orbit accuracy. Thus, satellites in either high altitude or highly elliptic orbits can maintain accurate position information.

Gyro Role

A small pointing satellite requires attitude determination capability at any orientation in space. Furthermore, the ability to measure rotation rates will be crucial to achieving stable conditions at any desired attitude. The micromechanical gyro data is optimally blended with the GPS data, thereby taking advantage of the gyro's ability to measure high-frequency dynamics and the GPS's ability to bound the error growth due to gyro drift.

The ability to instantaneously measure via GPS all components of vehicle attitude can place severe requirements on spacecraft geometry and antenna mounting to ensure observability. Antennas must be mounted such that both antennas making up a baseline can observe the same satellite without masking problems. Furthermore, this condition must be met for more than one baseline and satellite. Using the micromechanical gyro package as a reference permits us to bridge small gaps in interferometer coverage in both a spatial and temporal sense. This can reduce the number of antennas required and mitigate problems arising from field-of-view masking due to factors such as solar arrays or other instrumentation.

Attitude pointing and stabilization require high-speed, low-noise measurements for effective control. GPS measurements alone (without the gyro) will be too noisy for effective attitude control if antenna baselines are short. However, small size can also mitigate problems from vehicle structural flexibility and carrier wave ambiguity. Although not planned for at this time, the gyro package can also be used to eliminate other error sources in the GPS attitude solution:

ATTITUDE DETERMINATION FOR POINTING

Attitude determination depends upon both GPS and gyro attitude sensors for precise attitude knowledge and as a reference for the attitude control system and imaging slew maneuver. A Kalman filter combines the GPS and gyro measurements to provide an optimal estimate of satellite attitude and to calibrate the rate gyros. The filter estimates both attitude error and gyro drift rate bias.

A typical small imaging satellite tracks the nadir direction which nominally pitches at a constant orbital rate. For this reason, many of the gyro error sources appear to be equivalent bias drifts. For example, pitch axis scale factor cannot be separated from pitch axis bias drift. Similarly, roll and yaw axis gyro misalignments, which cause the orbital pitch rate to be projected into the roll and yaw axes, cannot be separated from bias drift in those axes. Therefore, without additional spacecraft maneuvers (which are not currently planned), the filter will estimate an equivalent bias drift for each orthogonal gyro axis.

Controlling satellite attitude is a three-axis problem but, with a three-axis stabilized vehicle, the problem can be treated as a set of three essentially uncoupled stabilization problems; therefore, only the pitch axis attitude determination problem is considered herein. The satellite is considered to be perfectly tracking nadir, pitching at a constant orbital rate. In reality the gyros, compensated by the bias drift estimate, couple the filter dynamics into the attitude control system dynamics thereby perturbing the satellite attitude.

The GPS attitude error is modeled as a 1st-order Gauss-Markov process. This model is based upon ground testing of the JAWSAT satellite mock-up by the AF Academy/University of Colorado. The GPS receiver used in that test was the Trimble TANS Vector and the antenna baselengths were approximately 50 cm. A similar experiment using the same receiver was recently conducted by Draper with very similar results. Most of the GPS attitude error is thought to be contributed by the effects of multipath reflections. The correlation time observed in the ground tests was approximately 100 s and the GPS error magnitude was approximately 0.28 deg about the yaw axis (along the nadir direction). The error magnitudes about the pitch and roll axes were slightly less. The nominal values used in this simulation were 0.28 deg for magnitude, with a correlation time of 100 s.

A covariance simulation of gyro and GPS attitude measurements combined in an extended Kalman filter was used to predict satellite attitude determination performance. Only a single-axis analysis was performed but the results should be quite representative of a three-axes implementation. The nominal performance of a system which combines gyros and simulated GPS measurements from a Trimble TANS Vector GPS receiver was predicted for a range of gyro performance error parameters. The covariance error simulation used the gyro and GPS model parameters summarized in Table 1 below. Performance of the 3-axis (tactical grade) μmechanical gyro package is typified by the range of 1 to 10°/hr bias stability. Bias stability of 0.01°/hr is typical of a high-performance, inertial navigation-grade gyro.

Table 1. Nominal Error Model Parameters

Parameter	Value (1s)
Gyro	
--angle random walk	0.05 deg/$\sqrt{\text{hr}}$
--bias drift stability over 10-hr period	10, 1, 0.1, 0.01 deg/hr
GPS (Markov process)	
--magnitude	0.28 deg
--correlation time	100 s

Covariance analysis predictions are presented for the system with the error model parameters shown in Table 1. Figures 7 and 8 show the rms (i.e., the square root of the covariance) attitude and bias drift error predictions as a function of the gyro error parameters.

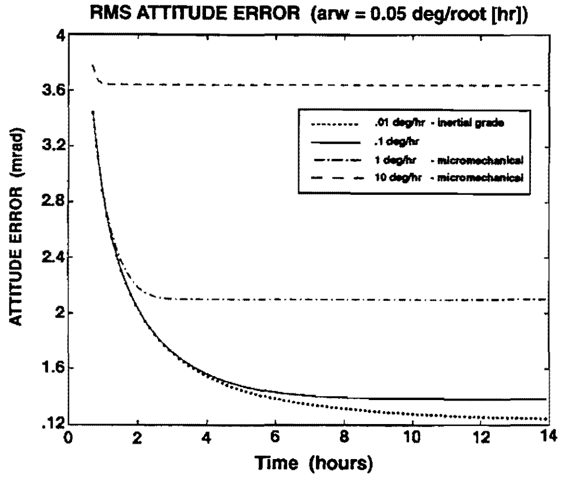

Figure 7. RMS attitude error prediction

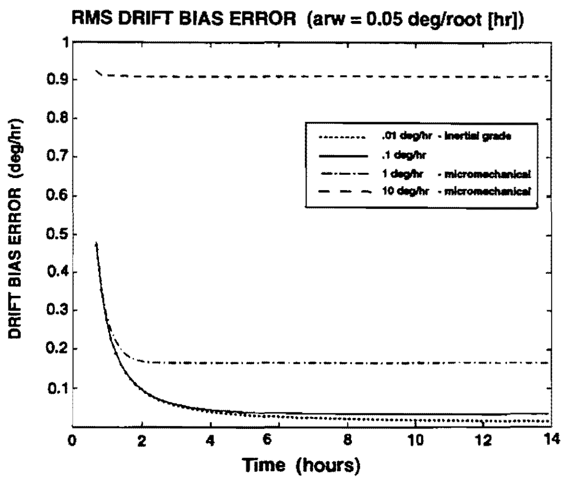

Figure 8. RMS drift bias error prediction

SUMMARY

A 3-axis μgyro package, comprising 3 orthogonally mounted single-degree-of-freedom silicon tuning fork gyroscope flat packs, has been described. These flat packs also contain the mixed-signal, application-specific integrated circuits (ASICs) for the gyro electronics. This package has been integrated with a GPS receiver as well as a guidance and navigation processor for terrestrial applications.

Analysis has shown the feasibility of an integrated μgyro/GPS system to control line of sight for a small pointing satellite. A ground-based hardware demonstration of the entire control system's capability to stabilize a telescope with reaction wheel actuators will be completed by early 1997.

BIBLIOGRAPHY

1. *Performance Analysis of a GPS Interferometric Attitude Determination System for a Gravity Gradient Stabilized Spacecraft*, J. Stoll, Draper Laboratory, MIT Masters Thesis, CSDL-T-1253.

2. *A Micromachined Comb Drive Tuning Fork Rate Gyroscope*, M. Weinberg, J. Bernstein, S. Cho, A. T. King, A. Kourepenis, and P. Maciel, Draper Laboratory, Proceedings of the ION 49th Annual Meeting, "Future Global Navigation and Guidance", June 21-23 1993.

3 *High Dynamic-Range Microdynamic Accelerometer Technology*, J. Sitomer and J. Connelly , Draper Laboratory, Proceedings of the ION 49th Annual Meeting, "Future Global Navigation and Guidance", June 21-23 1993.

4. *Micromechanical Sensors for Kinetic Energy Projectile Test and Evaluation*, A. Hooper, U. S. Army Yuma Proving Grounds, and R. Hopkins and P. Greiff, Draper Laboratory, U. S. Army TECOM Test Technology Symposium VII, March 29-31, 1994.

5. *Precision Strike Concepts Exploiting Relative GPS Techniques*, G. Schmidt and R. Setterlund, Draper Laboratory, NAVIGATION, Journal of the ION, Vol. 41, No. 1, Spring 1994, pp. 99-112.

6. *A Micromechanical Comb Drive Tuning Fork Gyroscope for Commercial Applications*, M. Weinberg, J. Bernstein, S. Cho, A. T. King, A. Kourepenis, P. Ward, and J. Sohn, Draper Laboratory, 2nd St. Petersburg International Conference on Gyroscopic Technology and Navigation, St. Petersburg, Russia, May 1995.

7. *A Micromechanical INS/GPS System for Guided Projectiles*, D. Gustafson, R. Hopkins, N. Barbour, J. Dowdle, 51st ION Annual Meeting, June 1995.

8. *An Innovative Bonding Technique for Optical Chips Using Solder Bumps That Eliminate Chip Positioning Adjustments*, T. Hayashi, IEEE Transactions on Components, Hybrids, and Manufacturing Technology, Vol. 15, No. 2, April 1992.

9. *Micro-electromechanical Instrument and Systems Development at Draper Laboratory*, J. Connelly, N. Barbour, J. Gilmore, M. Weinberg, 3rd St. Petersburg International Conference on Integrated Navigation Systems, St. Petersburg, Russia, May 1996.

10. *An Integrated GPS Attitude Determination System for JAWSAT*, B. C. Chesley and P. Axelrad, Proc. of the Institute of Navigation (ION) Conference, 1994.