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NASA's first in-space optical gyroscope
A technology experiment on the X-ray Timing Explorer spacecraft

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ABSTRACT

We describe a technology experiment on the X-ray Timing Explorer spacecraft to determine the feasibility of Interferometric Fiber Optic Gyroscopes for space flight navigation. The experiment consists of placing a medium grade fiber optic gyroscope in parallel with the spacecraft's inertial reference unit. The performance of the fiber optic gyroscope will be monitored and compared to the primary mechanical gyroscope's performance throughout the two-year mission life.

1. INTRODUCTION

Interferometric Fiber optic Gyroscopes (IFOGs) are rotation sensors used for commercial aircraft and cruise missile navigation.[1, 2] IFOGs are a small, solid state, light weight, low voltage, and low power alternative to mechanical gyros. These attributes make the IFOG ideal for space flight navigation use. IFOGs have the potential for > 15 year reliability. In addition, the IFOG measurement sensitivity can be scaled by adjusting the fiber loop diameter and/or the number of turns. As a result, IFOGs are capable of measuring incremental angles and rotation rates to a high level of accuracy.

NASA/Goddard Space Flight Center (NASA/GSFC) and Honeywell are teaming to fly an IFOG experiment on a NASA spacecraft. The X-ray Timing Explorer[3] (XTE) has been identified as the near term target of opportunity for the IFOG experiment. A closed-loop IFOG IRU (Inertial Reference Unit) will be placed on the XTE spacecraft as a separate, but distinct experiment.

The goal is to transfer the IFOG technology from aircraft and missile applications to routine NASA space flight use. This offers great benefits. NASA obtains a more cost efficient, reliable technology for space flight navigation with a number of proven suppliers and the navigation instrument market is enhanced and expanded for military and commercial suppliers.

The main objective of the XTE IFOG experiment is to prove the long term feasibility and reliability of the IFOG technology for space navigation. Other objectives are to: (1) establish a medium performance space qualified IFOG IRU specification for the NASA/GSFC Small Explorer (SMEX), Earth probe, and Explorer class spacecraft such as XTE, Tropical Rainfall Measuring Mission (TRMM), Far Ultraviolet Spectroscopic Explorer (FUSE,) etc., (2) provide flight heritage for a vendor-

supported optical guidance and control component for future spacecraft, (3) increase reliability of NASA spacecraft to support several medium and small class science missions, and (4) develop low cost, reliable on-shore market suppliers to support NASA missions.

We will discuss in the following sections the XTE project, science objectives, and pointing requirements; IFOG experiment; space-based and ground-based data processing; IFOG-to-XTE interface; and IFOG description and performance.

2. X-RAY TIMING EXPLORER

A Delta II will be launched from Kennedy Space Center (KSC) and carry the XTE into a 500 km to 600 km orbit with a 23 degree inclination. The XTE will be launched no later than April 1996 with a launch goal of August 1995. The mission life is anticipated to be greater than two years. XTE will weigh less than 7200 pound and use less than 800 Watts. XTE's three primary instruments are the All Sky Monitor (ASM), Proportional Counter Array (PCA), and High Energy X-ray Timing Experiment (HEXTE.) The ASM is being developed at Massachusetts Institute of Technology, the PCA at NASA/GSFC, and the HEXTE at University of California. XTE's science objective is to monitor and study 2 keV to 200 keV energy regions in the sky. The spacecraft must provide 0.1 degree pointing to any celestial source, 0.036 degree pointing knowledge, and less than 0.0083 degree jitter (3 sigma.)

3. IFOG IRU EXPERIMENT

The proposed experiment will compare the performance of the IFOG with that of the XTE's primary gyroscope, a mechanical gyroscope, for at least 2 years. By comparing the primary gyro's determined attitude with the IFOG's determined attitude, the IFOG's long-term reliability can be measured. The XTE On-Board Computer (OBC) and GSFC ground-based computers determine the spacecraft attitude from the XTE primary IRU data. However, only the GSFC ground-based computers will be used for spacecraft attitude determination using the IFOG IRU data.

4. DATA PROCESSING

Compensated incremental angle data from the IFOG IRU and primary mechanical gyro based IRU will be read by the XTE on-board computer at a 4 Hz rate. IFOG voltages, temperatures, and status will be read at a 0.5 Hz rate. The IFOG IRU will be read immediately before the primary gyro. The data flow for the two gyros will be different. In particular, the position or incremental angle data flow for both gyros is described in the following paragraphs.

The IFOG IRU data packet will be dumped to the Spacecraft Command and Data Handling Software for formatting. The packet format adheres to the standards set forth by the Consultative Committee for Space Data Systems (CCSDS.) After the IFOG IRU data is formatted it will be telemetered to the GSFC Flight Operation Team (FOT.) The FOT is responsible for retrieving all XTE telemetry, performing trending analysis (i.e. voltage vs. time), transferring data to the Flight Dynamics Facility (FDF), and data storage. The FDF is responsible for On-board Computer (OBC) Attitude Determination and Validation (AD&V.) During each XTE OBC AD&V, the FDF will measure residual statistics - a process that defines vectors and calculates vector differences for the primary gyro and IFOG relative to the star sensor. Also, the FDF will compare the IFOG determined attitude with primary gyro determined attitude. The FDF attitude determination process is similar to that used by the XTE Attitude Control System (ACS) OBC. Using the star tracker and gyroscope input, a Kalman filter determines an error and propagates a new spacecraft attitude.

Unlike the IFOG's data processing, the XTE Attitude Control System On-Board Computer will use the primary gyro data to determine the spacecraft's attitude. The XTE's star sensor data and the primary gyro's incremental angle data will be input to a Kalman filter. The filter is implemented with the XTE OBC. The OBC will determine the delta quaternion from which the new attitude is propagated. The attitude, star sensor data, and incremental angle are among the many types of data that are telemetered to earth.

5. IFOG INTERFACE

The IFOG IRU will be located on the top inside panel on the spacecraft side (see figure 1 and 2.) The IFOG is less than 10 pounds and is 7" wide, 7" high, and 10" deep. The thermal environment is between -10°C and +40°C. No thermal heaters are necessary. The XTE has three MIL-STD 1773 buses: the spacecraft, instrument, and ACS. The IFOG will interface to the ACS's bus. The Non-essential Power Bus will supply +28 V DC to the IFOG IRU. The IFOG IRU will also have access to a +28 V pulse that can turn-on and turn-off its internal switch. The relay allows the supply voltage to reach the IFOG IRU electronics. The IFOG IRU worst case power consumption is 30 Watts.

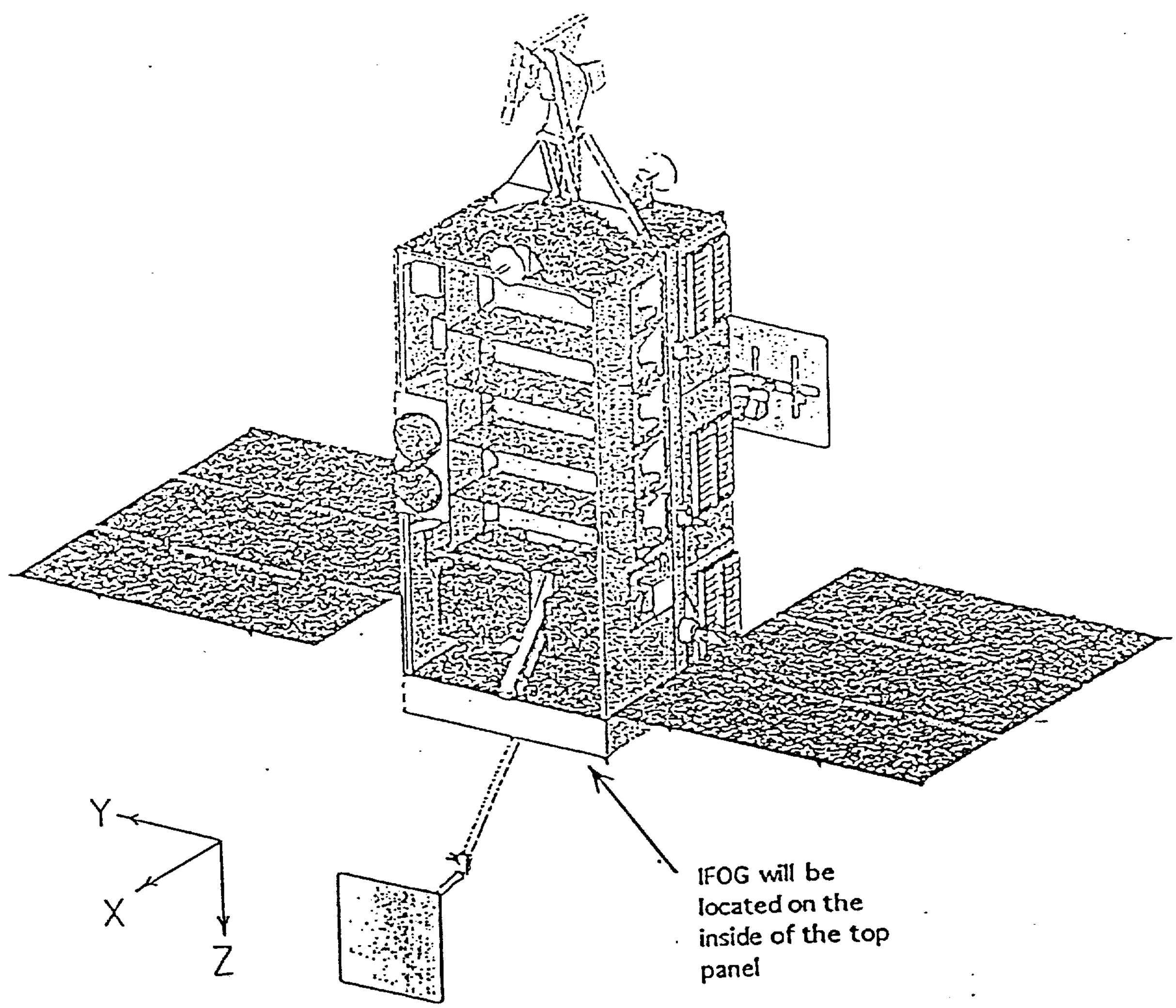

FIGURE 1 - XTE Deployed View

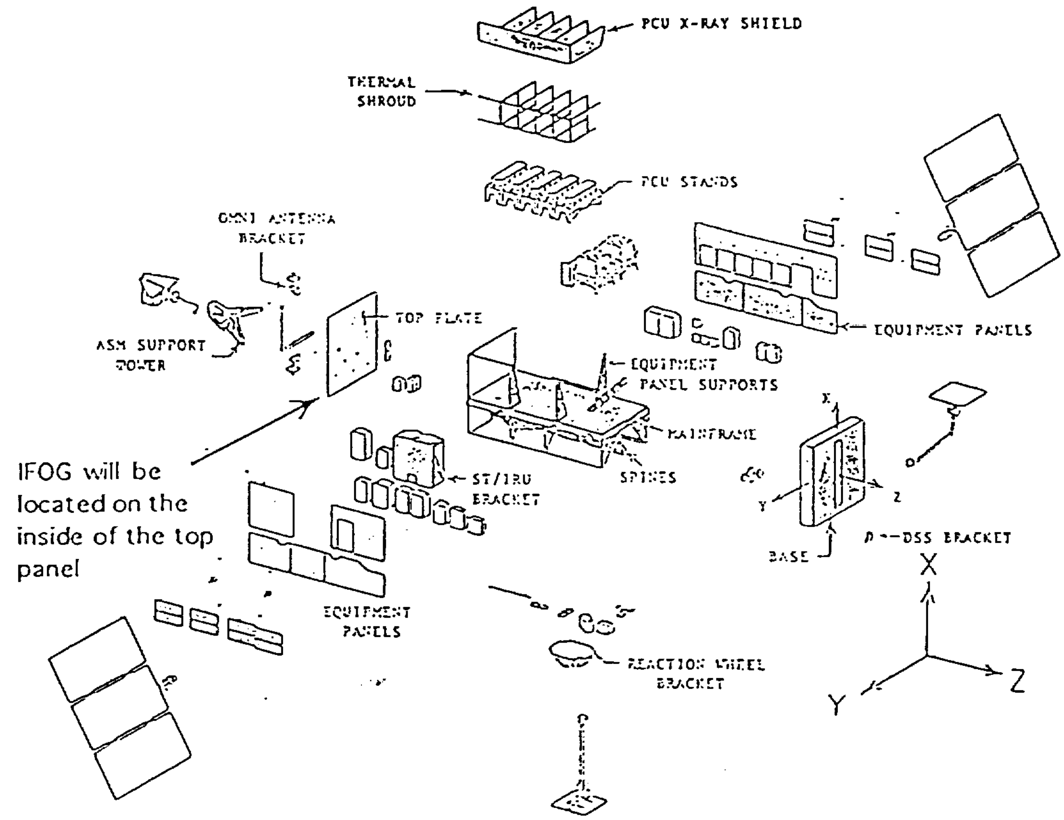

FIGURE 2 - XTE Exploded View

6. FIBER OPTIC GYROSCOPE DESCRIPTION AND PERFORMANCE

The XTE satellite IFOG IRU experiment uses three IFOGs mounted orthogonally on a sensor block assembly. Each of the IFOGs meets the following requirements:

- Bias stability — 0.01 deg/hr (1s)
- Scale factor uncertainty — <50 ppm
- Angular random walk — 0.005 deg/√hr
- Frequency response — >500 Hz
- Input axis alignment uncertainty— 100 μrad (1s)
- Readout resolution — 0.167 arc-sec/LSB
- Maximum angular rate — ±140 deg/sec
- Size –
  - Diameter — 3.6 in
  - Height — 1.1 in
- Weight — 300 gm.
- Full performance operational temperature range: -20 to +50 deg C
- Maximum temperature rate of change - 5 deg C/hour

As shown in figure 3, the IFOG consists of a solid state light source, a fiber-optic directional coupler, a Lithium Niobate ($LiNbO_3$) integrated optic multi-function chip (MFC), the fiber sensing coil, a photodetector and signal processing electronics. The sensing coil is roughly one kilometer in length, wound on a mandrel which is nominally three inches in diameter. The MFC consists of a polarizer, two phase modulators, and a "Y-junction" for splitting the source light into two waves of equal intensity. The two waves, after traversing the coil in the clockwise (cw) and counter clockwise (ccw) directions, are then recombined at the "Y-junction" and then directed to the photodetector via a fused fiber-optic directional coupler. In the absence of rotation, the cw and ccw waves see identical path lengths inside the sensing coil, and therefore interfere constructively (maximum intensity) at the photodetector. In the presence of rotation, the two waves experience a path length difference proportional to the rotation rate.

To measure the small phase shifts due to rotation, a standard AC bias modulation technique is used. The optical phase difference between waves is modulated, and the photodetector output is synchronously detected. In closed loop operation, the demodulated signal serves as an error signal for driving the sensor back to its null condition. This is achieved by applying a phase ramp feedback signal to one of the phase modulators on the MFC, which in turn, introduces a phase difference into the loop equal and opposite to that caused by rotation. The feedback signal then provides a digital pulse train of frequency proportional to rotation rate and an integrated output proportional to the accumulated angle of rotation (0.167 arc second/pulse).

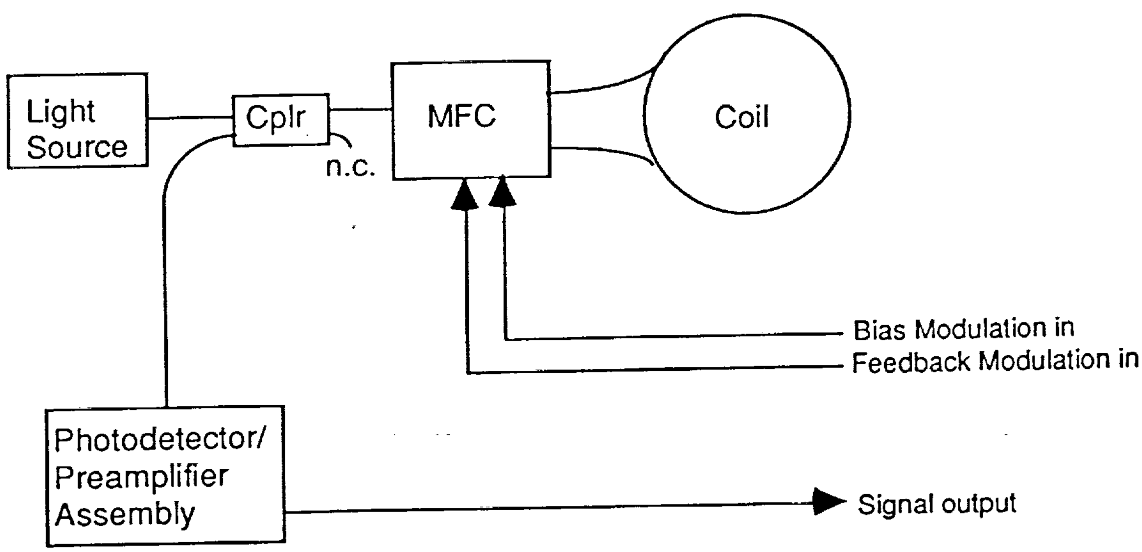

FIGURE 3 - IFOG Concept

7. IFOG IRU ELECTRONICS

The major IRU system subassemblies, shown in the block diagram of figure 4, are:

- Inertial Sensor Assembly (ISA)
  – Fiber Optic Gyros (three)
- MIB and wiring harness
- IFOG Support Electronics assembly
- Built-In Test (BIT)/Control Electronics assembly

- Processor assembly
- I/O assembly
- Power Supply assembly
- Chassis

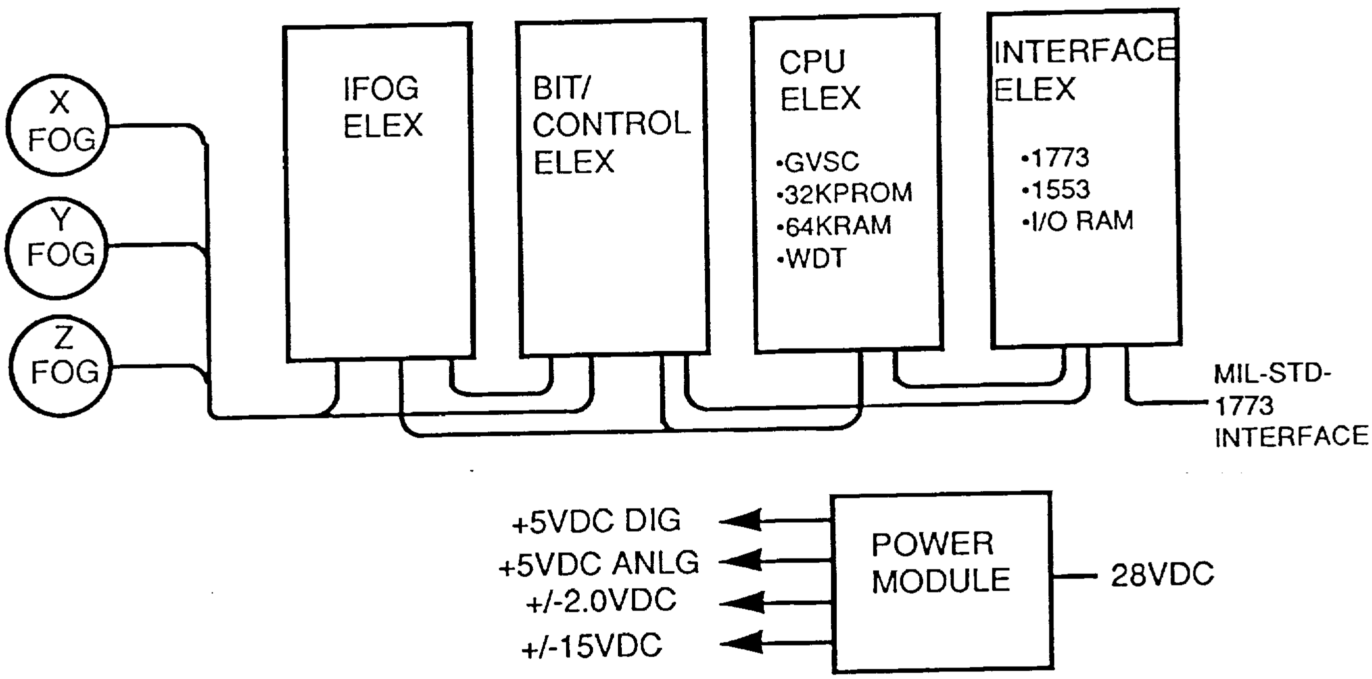

FIGURE 4 - Major IRU System Subassemblies

The IFOG Support Electronics assembly contains the closed-loop electronics for all three sensors. Each channel contains the bias modulation and control electronics required to close the loop. The assembly outputs two pulse streams for each gyro — one for a positive angular motion, and another for a negative angular motion about the input axis of the gyro. The pulses are then accumulated on the Built-In Test (BIT)/control electronics assembly.

The BIT/control electronics assembly contains pulse accumulators for counting and storing pulses from the gyro electronics. The assembly also contains a 12 bit analog-to-digital converter system for measuring temperatures and voltages. Temperatures are used for compensating the raw pulse counts and for BIT purposes. The voltages are for monitoring purposes as a part of the BIT.

The IRU system contains the Honeywell radiation-hardened RH1750 processor — a MIL-STD-1750A processor used in spaceborne computer systems designed by Honeywell. This processor contains floating point capability, and a has a throughput capacity of 2.0 MIPS for a DAIS instruction mix using a 14 MHz clock source. The processor currently has more than 50 percent spare throughput capacity, and more than 50 percent spare memory.

The processor assembly contains 32K x 16 bits of bipolar PROM that is downloaded at power-up to higher-speed RAM. The assembly contains 64K x 48 bits of static high-speed RAM. The RH1750 stores data into 32 bit locations, with additional bits used for Error Detection and Correction (EDAC) and column sparing. The bipolar PROM is powered down after downloading, conserving power. A virtual console port is also available for software development and test. A watchdog timer is used as a sanity check of the processor.

The I/O assembly contains a MIL-STD-1553B interface memory-mapped into the processor's memory. The assembly contains conversion circuitry for the XTE application to output or receive modified signals required for the MIL-STD-1773 receivers and transmitters.

Fiber-optic transmitters and receivers are contained in chassis-mounted connectors (SMA 905 type) manufactured by Honeywell and approved by NASA (see NASA Technical Paper 3227). The I/O assembly contains a discrete input that permits test equipment to enable either the 1553 interface or the 1773 interface. This option is used to allow the test configuration to be optimized because rate-table testing is difficult and expensive to implement with fiber-optic slip rings.

The power module contains only low-voltage power. The spacecraft primary 28 V DC is converted to the following voltages required by the IRU system:

- +5.0 VDC (digital electronics)
- +5.0 VDC (analog electronics)
- +15 VDC
- -15 VDC
- +2.0 VDC
- -2.0 VDC.

The power module operates per MIL-STD-1539, and is capable of operating from two independent power sources. Input power is filtered for EMI per MIL-STD-461C.

8. CONCLUSION

We have presented the details of a technology experiment planned for the X-ray Timing Explorer space mission. The experiment consists of placing a medium grade fiber optic gyroscope in parallel with the spacecraft's inertial reference unit. The performance of the fiber optic gyroscope will be monitored and compared to the primary mechanical gyroscope's performance throughout the two-year mission life. We anticipate that the successful completion of this technology experiment will greatly assist in the technology transfer of the fiber optic gyroscope and other photonic technologies to space applications.

9. ACKNOWLEDGMENTS

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