THE TAOS/STEP SATELLITE

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Abstract

The Technology for Autonomous Operational Survivability / Space Test Experiments Platform (TAOS/STEP) satellite was launched on a Taurus booster from Vandenberg Air Force Base into a nearly circular, 105 degree inclined orbit on March 13, 1994. The purpose of this satellite is twofold: 1) to test a new concept in multiple procurements of fast-track modular satellites and 2) to test a suite of Air Force Phillips Laboratory payloads in space. The TAOS payloads include the Microcosm Autonomous Navigation System (MANS), two Barnes Engineering Dual Cone Scanners, two Honeywell Generic VHSIC Spaceborne Computers, a Rockwell Global Positioning System receiver, a MIL-STD 1553B data bus, two laser sensors and a radar sensor. The payload experimentation period is about one and a half years and has been largely successful in spite of the failure of the STEP satellite's Inertial Measurement Unit, and, on the TAOS payload, one of the Generic VHSIC Spaceborne Computers, and the RF segment of the GPS receiver.

I. Introduction

The purpose of this paper is to describe the Technology for Autonomous Operational Survivability/Space Test Experiment Platform (TAOS/STEP) satellite jointly developed by the Air Force Space and Missile Systems Center and the Phillips Laboratory. The paper will also provide some of the lessons learned during system development and orbital operations. TAOS/STEP\textsuperscript{1,2} and DARPASAT\textsuperscript{3} were launched together on a Taurus\textsuperscript{4} booster from Vandenberg Air Force Base on March 13, 1994. The TAOS/STEP orbit is near circular at 300 nautical miles altitude and 105 degrees inclination.

The TAOS/STEP satellite uses the first Space Test Experiment Platform (STEP) spacecraft and is also referred to as STEP Mission 0. STEP is part of the Space Test Program (STP) of Space and Missile Systems Center Test and Evaluation Directorate (SMC/TE). TRW is under contract to build the generic modular STEP spacecraft buses. The STEP satellites carry new technology experiments developed by DOD laboratories and other DOD agencies. These satellites weigh from 400 to 1100 pounds with TAOS/STEP at the high end at 1050 pounds.
TAOS/STEP is a large "small satellite" at over half a ton. There are some big satellite attributes in the partial redundancy of the spacecraft and payload computers and in some of the attitude control system hardware. However, the development of the STEP satellite bus followed principles of fast-track, low-cost and streamlined testing. The time from turn-on of the STEP contractor (TRW) to launch was 48 months. The time for development, integration, and checkout of the TAOS payload to delivery to TRW for integration into the STEP satellite bus was 28 months.

The TAOS/STEP satellite during Integration at Vandenberg Air Force Base. The Satellite is Positioned in the Taurus Interface Structure.

The time from the beginning of TAOS payload integration into the STEP spacecraft until launch was 20 months. The cost of the TAOS/STEP satellite, excluding launch costs, was approximately $60 million dollars.

The STEP satellites are 12-sided cylinders. The cylindrical structure is 37.5 inches wide and (for the STEP/TAOS version) 76 inches high (see Figure 1). TAOS/STEP has its symmetry axis along the vector to the center of the Earth. All but one of the TAOS payloads are mounted on either side of a nadir-facing circular disc called the payload deployment module (see Figure 2). This module is a 12-sided, one half inch thick aluminum honeycomb. The Global Positioning System (GPS) receiver is on the spacecraft core module and its antenna is on the zenith facing surface of the satellite. The weight of the panel and the payloads is 223 pounds. When all the payloads are operating they consume about 160 watts.

The Air Force Phillips Laboratory developed most of the payloads and Phillips Laboratory Space Experiments Directorate (PL/SX) was responsible for the integration of the individual payloads into the TAOS payload. All of the payloads except for the GPS receiver were integrated and tested by GTE Government Systems in Mountain View, CA. The GPS receiver was integrated as part of the spacecraft by Defense Systems, Inc. (DSI, now CTA Space Systems) under contract to TRW in McLean, VA. The payload deployment panel was integrated with the spacecraft at DSI. Environmental tests at the satellite level were performed at NASA Goddard in Greenbelt, MD. The preliminary compatibility test with the
This module contains the spacecraft computers, the mass memory unit, the attitude determination and control hardware, and the telemetry equipment.

The electrical power system has 41 square feet of solar array area with silicon cells. These can provide 195 watts orbit average power. 100 watts of this is available on the average to the payloads, though peak values can be higher. The two solar panels were deployed to the sides of the cylinder after launch and have a wing span of 216 inches. Before the IMU failure the solar arrays were oriented toward the Sun by articulating the arrays about the support axis and yaw steering the satellite toward the Sun.

The solar cells provide power to charge 8 batteries. Each battery has 21 D size commercial NiCd cells and has a capacity of 4 ampere hours at the end-of-life (EOL). The cells are carefully selected and tested.

II. STEP Spacecraft
The spacecraft core module is another section parallel to the deployment module.

The Attitude Determination and Control Subsystem (ADACS) supports Normal
Mode, Attitude Maneuvering Mode and Safe Hold Mode. TAOS/STEP is 3 axis stabilized and nominally nadir pointing. The ADACS is required to point the satellite's symmetry axis toward nadir with an accuracy of plus or minus 0.5 degrees. It is also required to determine the spacecraft attitude about each axis with an accuracy of plus or minus 0.4 degrees.

The ADACS hardware complement consists of:

- 80C86-based dual redundant microcomputer
- 2 Ithaco T-Scanwheels, 2 reaction wheels
- Kearfott Inertial Measurement Unit with two 2-axis gyroscopes and 3 accelerometers
- Nanotesla magnetometer
- 2 TRW fine Sun sensors
- 3 Ithaco Torqrods
- 4 MRE-4 ACS thrusters, 2 Delta-V thrusters
- 2 solar array drive assemblies

The Inertial Measurement Unit (IMU) failed on July 21, 1994. Without the IMU data input, the attitude control system failed over from 3-axis control into Safe Hold Mode where the vehicle was in a controlled tumble and all unnecessary power loads were automatically shed. Due to the low solar beta angle (the angle between the Sun and the orbit plane) at the time, and despite the minimum power configuration, the batteries discharged and the satellite went into an under-voltage condition. All contact was lost with the satellite for two days. TRW was able to develop an approach to re-establish communications with the vehicle and rewrite the Safe Hold mode control software to maximize power generation. The spacecraft was brought back from the under-voltage condition just prior to the point where irreversible damage to the NiCd batteries would have occurred. After the satellite was in a long term safe configuration, work commenced on modifications to the attitude control software which would allow 3-axis control to be implemented without the IMU.

Making maximum use of the original attitude control system software and the remaining STEP ADACS hardware, TRW personnel developed and ground tested the new control algorithms. The scan wheels continue to provide infrared Earth sensor data. By measuring the chord during a scanwheel pass across the Earth, roll and pitch are estimated. But without the IMU, yaw updates are no longer available to the software. To regain 3-axis control, the yaw axis was fixed with regard to the orbit plane through the introduction of a momentum bias to the control wheels (which include the 2 reaction wheels and the 2 scan wheels). The magnitude of the momentum bias was estimated with ground based modeling.

At this point, TRW personnel made use of onboard attitude control software written for a Light-weight Reaction Wheel (LRW) being developed by TRW for DARPA but not manifested on TAOS/STEP. The momentum bias was introduced into the system by writing the bias value into memory location where the LRW bias
would have been stored. The attitude control system then used the existing code to process the bias. After a TAOS/STEP pass, the yaw axis control performance is validated using both the spacecraft Sun sensor and the magnetometer data.

Due to power generation issues related to solar beta angles in the TAOS/STEP orbit, two separate momentum bias control laws had to be developed. These control laws approximate the original yaw steering that maximized the solar array power. The first control law is a pitch momentum bias control system used when the satellite beta angle is less than 30 degrees. In this case the solar arrays are oriented perpendicular to the orbit plane. The second is a roll momentum bias control law used when the beta angle is greater than 30 degrees. In this case the solar arrays are oriented parallel to the orbit plane. The pitch momentum bias control software was successfully uploaded to the satellite in September, 1994, and payload operations resumed immediately. The roll momentum bias control was implemented in October.

The STEP Control and Data Handling (CDH) processor accepts commands from the ground and initiates the MIL-STD-1553B data bus traffic. On ground command, the CDH passes bus control to the Payload Executive in the payload computer. Once per minute, the Payload Executive passes bus control back to the CDH so that the CDH can resynchronize the clock on the payload computer. If both of the payload computers fail, the CDH can operate as bus controller to move data from the payloads to the Mass Memory Unit. The CDH uses a dual redundant 80C86 microprocessor operating at 6 megahertz. The Clock Generator Unit, the main onboard time source, is part of the CDH.

The STEP Mass Memory Unit (MMU), a remote terminal on the 1553B data bus, provides storage for the payloads and health and status data that is to be transmitted to the ground. The payloads can store up to 28 Megabytes of 1553B data between passes. The memory is partitioned into 64 kilobyte blocks to provide tolerance of memory part failures. On start up, the built-in test creates a map between logical and physical memory, so that the existence of unused memory blocks is transparent to the payload user.

The STEP telemetry uses the Space Ground Link System (SGLS) operating at S-Band. Spacecraft State-of-Health (SOH) and summary engineering data are sent down on SGLS Channel 18 (2.2875 GHz) at 32 kilobits per second. The payload data, detailed engineering, and command history data are all sent down from the MMU at one megabit per second on a subcarrier of Channel 18. The command uplink is on SGLS Uplink Channel 18 (1.831787 GHz) at a bit rate of one kilobit per second. Uplinks and downlinks are encrypted. The SGLS transponder has a pseudo random noise (PRN) mode so that satellite ranging can be performed when payload
data are not being transmitted.

III. TAOS Payloads

A. Generic VHSIC Spaceborne Computers

One of the most significant TAOS experiments is the operation of two Honeywell Generic VHSIC Spaceborne Computers\(^5\) (GVSCs). They employ the MIL-STD-1750A 16-bit instruction set architecture and achieve 2.5 Million Instructions Per Second (MIPS) while executing at 13.5 megahertz. The data interface of the TAOS GVSCs is the 1553B data bus. Each computer is 7.0" x 6.5" x 6.9", weighs 10.6 pounds, and consumes about 30 watts.

One GVSC is dedicated to executing the Payload Executive, which commands the MIL-STD 1553B data bus. Two copies of the Payload Executive are stored in non-volatile memory on each of the two GVSCs. The other GVSC is dedicated to executing the Microcosm Autonomous Navigation System (MANS). It is this latter computer which failed on December 4, 1994.

There is another application program which monitors computer problems and executes continuously in both computers. The GVSC Status Program (GSP) reports the clock speed, number of memory errors, and the RAM and ROM status.

The GVSCs contain one mega-word of static Random Access Memory (RAM). This memory has built-in Error Detection and Correction (EDAC) circuitry to detect all double bit errors and correct single bit errors. This feature is used to correct Single Event Upsets (SEUs). A scrubber program reads every word of RAM and when it finds an error in a word, it rewrites the word correctly. The scrubber frequency has been chosen so that it is very unlikely that a second SEU will occur.

*FIGURE 3. TAOS 1553B DATA BUS ARCHITECTURE SHOWING REMOTE TERMINAL NUMBERS.*
in the same word before the first error is corrected.

B. MIL-STD 1553B Data Bus

Another of the TAOS experiments is the connection of the payloads to one another and to the spacecraft with a 1553B data bus (see Figure 3). Many years ago the Air Force developed this standard to move data around aircraft, so there is a great selection of hardware and test equipment. The TAOS/STEP experience indicates that the 1553B data bus is an excellent interface method for satellite payloads with a modest data rate. The bit rate on the bus is 1 megahertz with overhead in the hardware and the packets. The maximum length of a 1553B data message is 32 16-bit words.

There are three types of 1553B bus entities: 1) the bus controller; 2) remote terminals; and 3) bus monitors. In normal operations one GVSC is the bus controller. If both payload computers fail, the spacecraft CDH will become bus controller. The second GVSC, each Dual Cone Scanner (DCS), and the Mass Memory Unit are remote terminals.

There are also two processors: the Bus Interface Module (BIM) and the Bus Interface Unit (BIU), which were developed for the sole purpose of providing the remote terminal functions for hardware that did not have a 1553B interface. The BIM connects the laser and radar sensors to the bus and the BIU provides the attitude control system data from the spacecraft bus to MANS via the 1553B data bus.

Bus monitors do not change the bus message traffic. They just listen and record. The Mass Memory Unit can function as a bus monitor and record all the bus traffic for later transmission to the ground. Bus monitors played a crucial role in the TAOS ground testing methodology.

One major advantage in the use of the 1553B data bus is the operational flexibility it provides. The normal traffic on the bus is determined by a table of sources and destinations of messages, and the frequency of these messages. The TAOS Payload Executive has nineteen tables resident in it. If none of these nineteen tables is appropriate for a particular application, then a new table can be transmitted to the Payload Executive from the ground. Therefore, the entire pattern of onboard message flow is data driven.

As an example, when the MANS GVSC failed, bus tables were created which sent the Dual Cone Scanner data directly to the Mass Memory Unit.

The use of a 1553B data bus necessitates bus controller software. The Payload Executive is a small Ada program with only 3,000 lines of Ada source code. Nevertheless, problems with this program during the payload integration phase caused its delivery to move onto the critical path and stay there for 6 months. A very complete test procedure was developed for the Payload Executive. During
the first test of the entire Payload Executive there were 65 Problem Reports. A major debugging effort ensued and the second set of tests produced 34 Problem Reports.

For a while it looked as if the Payload Executive would need to be rewritten from scratch but this was not done and eventually all the problems were fixed. In retrospect, the problems arose because the coding of the Payload Executive commenced before a comprehensive top level design was achieved. After leaving the GTE facility in Mountain View, CA, only 3 program errors were found and the program has executed flawlessly during the remainder of ground-based integrated system tests and during on-orbit operation.

C. Microcosm Autonomous Navigation System (MANS)

MANS has been discussed elsewhere\textsuperscript{6,7} at length. Microcosm in Torrance, CA, introduced MANS as a Small Business Innovative Research (SBIR) effort. Its principle of operation is that two scanners can provide time-tagged locations of the edges of the Sun, Earth, and Moon and use these data to determine the position, velocity, and attitude of the satellite. The MANS Ada source code is approximately 35,000 lines of code.

The MANS hardware consists of a single computer (one of the GVSCs discussed above) and two Dual Cone Scanners\textsuperscript{8} built by Barnes Engineering. Each scanner head weighs 3.4 pounds. Each scanner has an 8.8 pound electronics box which converts the analog scanner signals to 1553B output. A scanner and its electronics box consume 14 watts after the scanner reaches its steady-state operation. There is a half second start-up drain of 23 watts as the scanner spins up.

The MANS experiment placed constraints on the construction and testing of the satellite. The Dual Cone Scanners are precision optical devices so that cleanliness must be maintained when their covers are off. Also, the satellite structure must be stiff enough so that the principal attitude axes maintain the same relationship to the scanner axes. This relationship must be carefully measured before launch. This measurement necessitates an alignment cube on each of the Dual Cone Scanners and the attitude control equipment. When this measurement was performed on TAOS/STEP, it was done first with the satellite in its standard upright position and then with the satellite inverted to provide an estimate of the alignment angles in the zero gravity environment.

The scanners measure the edge of the Earth with an infrared sensor and the edges of the Sun and Moon with a visible sensor. The Sun signal is unambiguous since it saturates the visible sensor. The Moon signal is not only less intense but it changes throughout the month and is unavailable from the third quarter to the
first quarter.

Logic in the electronics box sets a Moon triggering threshold at one half the maximum Moon signal level in order to obtain the most accurate location of the edge of the Moon. The designers anticipated that the threshold setting logic could become contaminated by visible light reflected from the Earth so they put in a test to exclude the Earth on the basis of it exceeding the one half degree size of the Moon. Apparently the implementation of this part of the logic is incorrect because the Moon signal disappears when the satellite is over the Sun-illuminated Earth. The Microcosm contract with Barnes did not specify that the Moon signal not be contaminated by an Earth visible signal; therefore, this requirement was not tested. The practical effect of this problem is that the MANS software is deprived of the Moon signal during most of the time the satellite is in the sunlight. Therefore, there are almost no simultaneous Sun and Moon inputs to MANS. This turns out to be a major limitation of the TAOS demonstration of the MANS performance.

Since the failure of the GVSC running MANS, the ground equipment has been enhanced to model on-orbit processing with inputs from the Dual Cone Scanners. This methodology has been validated by comparing the results of MANS software operation on the ground with results computed on TAOS before the GVSC failed. The ground computations are performed on the Honeywell Prototype Development Unit and the spare GVSC flight computer. Microcosm is continuing the assessment of MANS operation. In addition, Phillips Laboratory has funded the Space Dynamics Laboratory of Utah State University to perform an independent assessment of MANS performance.

D. Global Positioning System Receiver

DARPA funded Rockwell in Anaheim, CA, to develop three GPS receivers for use in space. One is on TAOS and one is on DARPASAT. The receiver weighs 8 pounds and consumes 12 watts of power. It is 7.8" x 7.1" x 5.1".

The TAOS GPS receiver was integrated as an experiment and did not provide data to the STEP spacecraft ADACS. Also, GPS time was not made available to the payloads. Satellite time is based on the clock in the spacecraft CDH.

The GPS receivers have 6 independent RF and processing channels and thus can simultaneously process 6 GPS satellites. The 4 channels which provide the lowest Geometric Dilution of Precision (GDOP) are selected to calculate the navigation solution. The other two are processing and are ready to be swapped in to keep the GDOP minimized.

The receivers are not designed to mitigate the effects of the GPS Selective Availability which the Air Force employs to deliberately degrade the GPS accuracy. Therefore, it is expected that the accuracy
of these receivers should be 80 to 100 meters Spherical Error Probability (SEP). The GPS receiver sends 3 messages when it is in navigation mode. The navigation solution message contains the satellite position and velocity in Earth Centered Earth Fixed (ECEF) coordinates. The measurement data message contains the raw values of pseudorange and delta range between the receiver and the GPS satellites. The ephemeris message contains an estimate of the TAOS/STEP orbital parameters.

Rockwell tested the receiver with a rack of programmed transmitters called the Simulation and Evaluation System (SEVS). These transmitters inserted the correct Doppler shift between the simulated host satellite and the GPS satellites. These tests indicated that if Selective Availability is not corrupting the GPS signals and if the GDOP is less than 6, then the SEP would be 12 to 15 meters.

Personnel at the Test Support Complex at Space and Missile System Center, Detachment 2, and the Jet Propulsion Laboratory (JPL) studied the position accuracy of the TAOS GPS receiver 10. The analysts used the GPS receiver's data messages as an input to JPL's Multiple Interferometric Ranging Analysis and GPS Ensemble (MIRAGE) software. MIRAGE uses very sophisticated force models and has the ability to effectively remove in batch processing mode the Selective Availability accuracy degradation. The MIRAGE orbit positions have an error on the order of 2 meters, which is so much smaller than the TAOS GPS navigation solutions or the SGLS state vector determinations, that they can be used as truth data. The study showed that the root mean square position errors were approximately 58 meters for the TAOS GPS navigation solutions and 23 meters for the SGLS state vectors.

The GPS receiver required an initialization message, containing an estimate of the host satellite state vector, each time it was turned on. This information made it possible for the receiver to determine what GPS satellites were in view and reduced the frequency search space. The initialization message had to be accurate to plus or minus 150 kilometers in position, plus or minus 180 meters per second in velocity, and plus or minus one second in time.

During the summer of 1994, the GPS receiver became erratic. The cause was originally assumed to be problems with the initialization process or aging of the GPS satellite almanac the GPS receiver maintains. After careful testing, it was determined that the receiver was not entering navigation mode but spending all of its time in the acquisition mode of looking for a minimum of 4 GPS satellites. Rockwell personnel carefully examined the TAOS/STEP telemetry records. These records showed that some satellites were being acquired and that all 6 channels
were processing data. It was concluded that the carrier to noise level was at least 10 decibels lower than obtained in the tests shortly after launch.

It appears that the problem is with the antenna or the cable from the antenna to the receiver. The receiver was spending all of its time looking for stronger signals and not concentrating on overhead closer satellites that are inherently stronger in signal. A test was devised to test this conclusion. The test required that the GPS satellites be nearly overhead to maximize the carrier to noise ratio and that there only be 4 satellites so that precious search time would not be wasted on extraneous satellites. Rockwell found that there was a period of time when there would be only 4 satellites nearly overhead. The GPS receiver proceeded into navigation mode and stayed there until one of the 4 moved out of range. The receiver has not been tested since.

E. Sensor System

The TAOS payload has three sensors and a Bus Interface Module to connect them to the 1553B data bus. One of the two laser sensors on the satellite was built by Sandia National Laboratories of Albuquerque, NM, and the other was built by Intelligent Interactive Imagery Corporation of Foster City, CA. The radar sensor aboard TAOS was built by GTE Strategic Electronic Defense Division of Mountain View, CA.

The Bus Interface Module uses an 80C186 microprocessor operating at 10 Megahertz. It has 512 kilobytes of static RAM, 256 kilobytes of EEPROM for applications, and 64 kilobytes of EEPROM for the operating system. It communicates with the laser sensors via serial interfaces at 9600 baud and with the radar sensor via four custom serial lines operating at 250 kilobaud. This module is typical of the type of subsystem one can use to integrate payloads to the 1553B data bus which do not have their own 1553B circuitry.

FIGURE 4. TAOS AUTOMATIC TEST EQUIPMENT (ATE). The TAOS payload ATE screens with the integrated system test setup at Vandenberg Air Force Base. The VAX screens are on the left and the PCs are on the right.
IV. Integration and Test of the TAOS Payload

Payload integration took place at the GTE Mountain View, CA, facility. Mechanical fit checks and electrical correctness tests were performed as each payload was delivered. Then the more complex function of data compatibility was tested. GTE built an Automatic Test Equipment (ATE) based on a Digital Equipment Corporation VAX 4000, a VAX 3500, a VAX 3400 and a number of PCs.

The tests modeled the flight configuration as closely as possible by simulating missing hardware with a VAX or a PC. Each VAX had a 1553B interface board made by Digital Technology Incorporated (DTI). This board had a design flaw which caused the VAX to crash when the board throughput was high. This created a minimum 20 minute delay in testing during which time the VAX rebooted. The DTI board had some timing features described below which made it possible to meet the onboard time accuracy requirement.

A PC 1553B board made by Data Devices Corporation was used during every TAOS payload system test. This PC showed the cumulative bus errors for each remote terminal. System problems of every type often manifest themselves first on this PC screen.

Files were generated for each payload from the 1553B bus traffic (see Figure 4). The most significant parameters from these files were culled by the ATE Report Generators. These reports were displayed on the VAX screens. This made it possible to tell at a glance whether the Payload Executive was operating in the correct mode or whether there were any bus errors. Every payload's health and status output was summarized and available immediately.

V. Integration and Test of the TAOS Payload into the STEP Satellite

The TAOS payload ATE and its attendant Report Generators were used during every phase of the integration and test from individual payload integration all the way to launch site preparation at Vandenberg. During payload to spacecraft integration the payload operation was monitored without interfering with the spacecraft test personnel since only a 1553B cable needed to be attached to the 1553B data bus. The same method worked well during the environmental tests. In addition, the TAOS ATE is used the same way at the GTE Contractor Support Facility with data that has been telemetered from the satellite. The only place this ATE is not used is to support real time orbital operations. TRW of Colorado Springs, CO, developed the real time ground systems based on SUN workstations. The use of the TAOS payload ATE from beginning to end has certainly reduced development cost and risk.

After the payload deployment panel was integrated to the spacecraft at DSI, the integrated system tests were performed using both the TAOS ATE and STEP satellite ATE. The spacecraft was transported to NASA Goddard for the following tests:
- Electromagnetic Compatibility
- Thermal Vacuum and Thermal Balance
- Preliminary CSTC Compatibility
- Acoustic
- Pyro Shock
- Mass Properties Determination
- Alignment

The final integrated system tests and the final Det 2 Compatibility Test were performed at Vandenberg Air Force Base.

VI. Satellite Time

One of the TAOS payload requirements was that individual experiments have time accurate to 1.5 milliseconds. This was necessary to support the MANS performance. The Payload Executive broadcasts the time to all remote terminals every three seconds using the GVSCs internal clock. This clock is not accurate so it is reset every minute by passing bus control to the STEP satellite CDH. The CDH resets the GVSC clock, using the accurate time from the Clock Generator Unit, and then passes bus control back to the Payload Executive. The 1553B community frowns on this dynamic bus control because if the control is not returned properly there will be no bus controller and the bus will stop. The TAOS/STEP experience is that this process has never failed, either during integration and test or on orbit.

The onboard CDH Clock Generator Unit uses a temperature compensated oscillator. This clock drifts approximately 0.2 microseconds per second, therefore it must be reset from the ground. This process of resetting the clock means that time must be sent down and the error measured so that an adjustment command can be sent up. The error measurement process requires a knowledge of all delays from the time that the on-board clock is read until the error can be measured on the ground. The time of flight is just a few milliseconds, but hardware delays can be significantly longer.

It was decided to measure the total time delay from reading the onboard clock during the Det 2 compatibility test at Vandenberg. This test was made possible by the fact that system times are regularly sent over the 1553B data bus. The DTI card stamps all of the messages with the value of its 32 bit counter in microseconds. It was realized that if the value of the counter could be related to the UTC value of time it would be possible to calculate a relationship between the content of time messages and when they actually were sent.

The DTI card's internal clock has two limitations: 1) it drifts erratically; and 2) its absolute value has no meaning at all. But the card offers two redeeming features that bypass these limitations. The internal clock can be replaced by an external 1 Megahertz clock signal and the counter can be reset with a single external pulse. To take advantage of these features, a PC board with a very accurate 1 Megahertz clock signal, based on a GPS receiver, was purchased from KTS Odetics in El Paso,
The board was modified to send out a pulse at a prescribed time in the future. If the pulse was sent out at 1:00 p.m., the DTI counter output then equalled the number of microseconds from 1:00 p.m. There was no drift because of the GPS receiver.

During the Det 2 compatibility test all the on-board clock's messages were recorded and a linear equation was derived between this clock and UTC. It was then possible to measure the delay between the time this clock was read and its reception at Det 2. The delay was measured to be 526.4 milliseconds. This number is used, along with the effects due to range to the satellite, every time the onboard clock error is measured.

During a MANS test in October 1994, the clock error was measured 7 times in 5 days. A linear fit has been made between UTC and onboard time. The slope indicates the clock drift rate and the intercept is the clock error. The standard deviation of the individual measurements and this linear fit is .5 milliseconds. The size of this deviation is an indication that the methodology is successful. Any satellite with an onboard time accuracy requirement needs a method such as this or an accurate time source such as GPS.

Acknowledgments

Many people made the TAOS/STEP success possible but three members of the TRW team and four members of the GTE team were outstanding in overcoming the major TAOS/STEP problems encountered during this program, working weekends and long hours every day.

On the STEP team Mr. Mike French, the TRW site manager at Onizuka Air Force Base, provided unrelenting technical support implementing the successful recovery of the TAOS/STEP spacecraft IMU failure. The Attitude Control System engineers, Mr. John Sutila and Mr. Tom Morphopolous provided the outstanding and herculean technical creativity and engineering that made the recovery from the IMU failure possible.

On the TAOS Payload team, Mr. Jim Rinkert was the payload system engineer; he never let shortcuts reduce the system reliability. Ben Wilson developed the TAOS payload ATE and then ran all of the payload tests and ran them correctly. Jim Marcelli programmed the BIM and performed all the ad hoc coding needed during integration and test. Linden Laird developed the final version of the Payload Executive and did what the experts said was not possible, making it a very reliable program.

References


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