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MICROCOSM AUTONOMOUS NAVIGATION SYSTEM ON-ORBIT OPERATION

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This paper describes the operation of the Microcosm Autonomous Navigation System (MANS) which was launched as an Air Force Phillips Laboratory Technology for Autonomous Operational Survivability -(TAOS) experimental payload on March 13, 1994. MANS determines the position, velocity, attitude, attitude rate, ground look point, and lighting conditions of the host satellite. As implemented on TAOS, MANS uses two opposite facing horizon scanners from Barnes Engineering known as Dual Cone Scanners because of their two Earth limb locating infrared cones. The Barnes sensors also have two narrow slits admitting light to a detector in the visible band to locate the edges of the Sun and Moon. The MANS software, developed by Microcosm, runs on a Honeywell Generic VHSIC Spaceborne Computer (GVSC). This output consists of short term averaged results and very accurate Kalman Filter solutions. The paper describes how MANS was implemented on TAOS and presents results to date of the MANS on-orbit operation and evaluation.

INTRODUCTION

The Microcosm Autonomous Navigation System (MANS) was launched into space as part of the Space Test Experiment Platform (STEP) Mission Zero on March 13, 1994***. MANS is one of the Technology for Autonomous Operational Survivability (TAOS) payloads developed by Phillips Laboratory for the STEP Mission Zero spacecraft. The 1050 pound satellite was launched by a Taurus booster from Vandenberg Air Force Base into a 300 nautical mile high, nearly circular orbit inclined at 105 degrees.

The purpose of MANS is to supply accurate values of the host spacecraft position, velocity, attitude, attitude rate, ground look point, and lighting conditions with an inexpensive fully autonomous navigation system. There has been extensive research in satellite autonomous navigation, beginning in the 1960s[1]. Earlier papers have discussed the theoretical expectations of MANS[2,3,4]. This paper will address how MANS was implemented on the STEP spacecraft, the lessons learned, and some of the results to date.

The STEP Mission Zero spacecraft is nominally a cylinder with its symmetry axis along the vector to the center of the earth. Most of the payloads are mounted on a circular

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disc which is on the nadir-facing part of the cylinder. The Dual Cone Scanners (providing primary input to MANS) are attached to this disc, on opposite sides of the cylinder. Figure 1 is a sketch of the satellite.

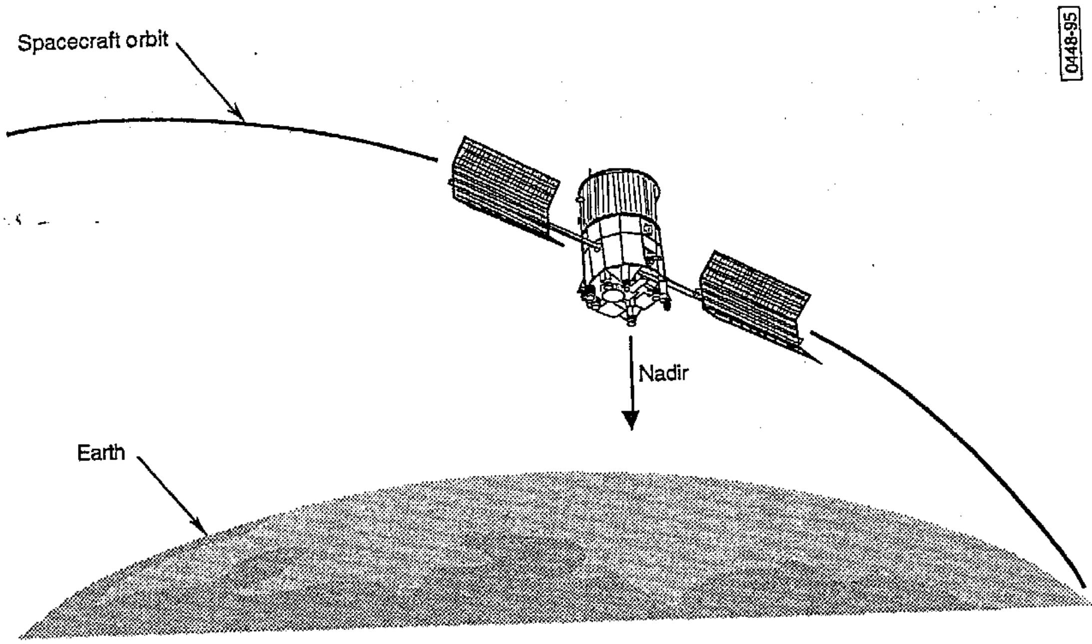

Figure 1. Sketch of TAOS satellite showing nadir pointing payloads.

As implemented on TAOS, MANS hardware consists of two scanners and their associated electronics units, the computer dedicated to running the main programs, and the required connectivity. The scanners have two visible slits and two infrared pencil beams. Since two infrared cones are generated in space this type of sensor is referred to as a Dual Cone Scanners, or DCS for short. The scanners rotate about their main axes at 4 rotations per second. The light entering the visible slits determines the location of the leading and trailing edges of the Sun and Moon. The infrared cones are used to determine the acquisition of the edge of the Earth and the loss of the Earth disc by observing the Earth's atmosphere in the 14 to 16 micrometer carbon dioxide absorption band. The electronics units preprocess the optical signals and provide the angles of the leading and trailing edges to the main program running in a general purpose computer.

The MANS software can use, but does not require, data from any of the following specific sensor types: Global Positioning System (GPS) receiver, Earth sensor, Sun and Moon sensor, gyroscope and accelerometer, and star sensor. With an accurate on-board clock, MANS can run indefinitely without external inputs (e.g., GPS signals). Except in a test mode, the software evaluates and uses the best available data at any point.

The high level software runs in a Generic VHSIC Spaceborne Computer (GVSC) built by Honeywe11[6]. This 2.5 Million Instructions Per Second (MIPS) computer executes the MIL-STD 1750A, 16 bit length, instruction set. This computer has one megaword of static random access memory, weighs 10.6 pounds, and consumes about 30 watts of power. The MANS software is divided into 3 main modules. The

“Deterministic” module provides data validation and processes the DCS data. The “Ephemeris” module contains the precise location of the Sun and Moon. The “Kalman” module filters the output of the Deterministic program.

Data flow from the scanners to the MANS GVSC and from this computer to the satellite telemetry downlink is over a MIL-STD 1553B data bus. A second GVSC executes the Payload Executive, which controls all the transfers over the 1553B bus. The 1553B protocol time multiplexes all sets of transfers. The bus traffic has been configured to have a one second major frame and a one eighth second minor frame. The primary MANS bus modes transfer DCS data to the MANS processor every minor frame. The 1553B bus is also used to transfer the MANS output to the Mass Memory Unit (MMU) and to provide time updates to all the payloads. Time is sent from the Payload Executive to the payloads every three seconds. The GVSC running the Payload Executive is updated by a temperature compensated spacecraft clock, the Clock Generator Unit (CGU), every minute. The CGU is updated from the ground as needed.

The early experimentation focused on the operation of the scanners. One scanner’s rotation axis is nominally parallel to and in the same direction as the solar al vector when the Sun is 90 degrees from nadir. This scanner observes the edges of the Sun in its visible channel and the Earth in the infrared channel. The scanner on the opposite side of the satellite views the Moon in its visible channel and the Earth in its infrared channel. There has been an unanticipated difficulty viewing the Moon when the satellite is illuminated by the Sun. Starting the Kalman filter has been a problem which has been overcome by uploading one of the changeable parameters in the MANS software.

PERFORMANCE REQUIREMENTS

The MANS performance state vector requirement is to determine the spacecraft’s position error with a three sigma error of 400 meters and to determine the velocity with a three sigma error of 0.4 meters per second. The attitude requirement is to determine the angle about each axis with a three sigma error of 0.05 degrees. The attitude rate requirement is to determine the angular rate about each axis with a three sigma error of 0.005 degrees per second.

There is also a set of goals. The position goal is to determine the position with a three sigma error of 100 meters. The velocity goal is a three sigma error of 0.1 meters per second. The attitude determination goal is to determine the angle about each axis with a three sigma error of 0.03 degrees. The attitude rate goal is the same as the requirement.

To achieve the MANS output requirements, the scanners must provide adequate angular accuracy. The Earth horizon crossing angles must be determined with a one sigma error of 0.1 degrees. The crossing angles of the Sun and the full Moon must have a one sigma error of 0.01 degrees. The error for the Moon at either quarter phase is increased by the lower signal to noise ratio and should have a one sigma error of 0.03 degrees.

Time is another important input parameter for MANS. The time delivered to MANS, or generated by the MANS computer, must have a one sigma error of 1.5 milliseconds.

There are other sources of system error. It has been assumed that it is possible to reduce these to negligible levels. An example of such an error is the angular misalignment of the two scanners. This error was estimated before launch and an alignment matrix was created. It is assumed that the values of the matrix can be chosen to further reduce the MANS error based on on-orbit scanner data.

SPACECRAFT SUPPORT OF MANS

The STEP Mission Zero spacecraft hardware supports the payload operations. The main spacecraft computer is the Command and Data Handler (CDH). Its primary function is to pass commands from the ground to the appropriate payload. Spacecraft time is maintained on the Clock Generation Unit, which is a CDH subsystem. As discussed in the section on MANS time input, the CDH resynchronizes the time on the Payload Executive GVSC once every minute.

The MMU provides on-orbit storage for payload data collected between ground contacts. The data are input to the MMU from the 1553B bus. Up to 28 megabytes may be stored. The most intense bus mode can **fill** the MMU in 105 minutes. The orbital period is 95 minutes so that almost one ground support per revolution is needed to support this mode. During ground supports the MMU is emptied at a rate of one million bits per second and when it is full it can be emptied in about 4 minutes.

CONNECTIVITY

The data from the scanners to the MANS software and the MANS output data to the Mass Memory Unit are transported over the MIL-STD 1553B data bus. This bus operates at one megabit per second. The bus is controlled by the Payload Executive. The 1553B data bus has two active types of participants, a single Bus Controller (the Payload Executive GVSC), and all the other nodes on the bus, referred to as Remote Terminals. Each scanner is a Remote Terminal along with the MANS GVSC and the Mass Memory Unit. Figure 2 depicts the raw data flow from the scanners to the MANS GVSC and the flow of the navigation data from the MANS GVSC to the MMU in preparation for transmission to the ground.

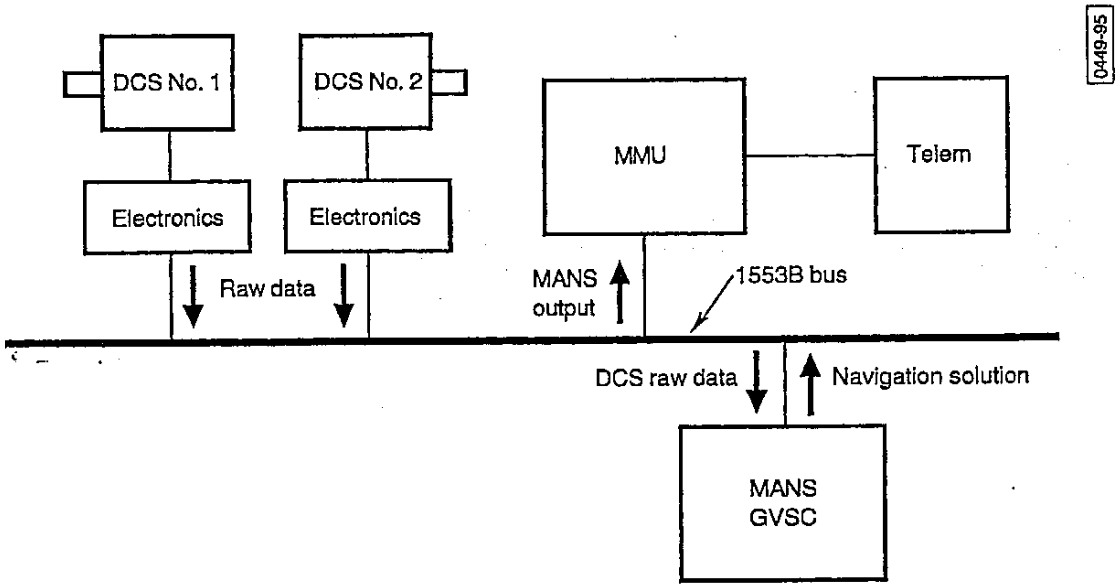

Figure 2. This figure shows the flow of data from the Dual Cone Scanners to the MANS Generic VHSIC Spaceborne Computer and from this computer to the Mass Memory Unit for eventual transmission to the ground.

The Payload Executive uses a built-in table to determine which transfers take place during each minor frame. For each transfer the table lists the source, the destination, and the number of words to be transferred. Up to 32 words can be transmitted during a transfer. The definition of the source and destination includes the number assigned to the particular Remote Terminals involved in the transfer and the respective subaddresses. Each Remote Terminal has 32 transmit subaddresses and 32 receive subaddresses. The precise data to be fetched from a source must be placed in the appropriate subaddress by the programming of that Remote Terminal.

There are 19 bus mode tables of which 14 are used to transfer DCS data to the MANS software. The particular bus mode to be used is selected based on the data which must be moved between payloads or to be sent to the ground. Another consideration in bus mode selection is the amount of data generated. It is possible to arrange one support per orbital revolution, but it is very difficult and can only be done for about four orbits during an off prime shift. Therefore bus modes which will only fill the MMU in 3 orbits or more are preferred.

There are two copies of the Payload Executive in EEPROM in the Payload Executive GVSC. The original design provided that if this GVSC failed, the Payload Executive could be run in the other GVSC to bring the raw Dual Cone Scanner data to the ground. In this case it would not be possible to run MANS on orbit, but at least the actual scanner data could be entered into the MANS software running on the ground. If both the GVSCs fail, there are contingency bus modes where the CDH is the Bus Controller to bring the scanner data to the MMU.

The 1553B data bus provides built-in redundancy by having two physical cables. One is for the primary A channel and the other is for the back up B channel. The Payload Executive starts out using the primary channel and can switch to the B channel whenever there is a transmission failure.

If the bus mode tables resident in the Payload Executive GVSC do not meet the experimenter's needs it is possible to design and upload a custom table. This feature has been used to good advantage with MANS when the mode usually used demanded too many ground supports but did not supply needed internal software parameters. A new mode was designed and used successfully which only filled the MMU in two and a half orbital revolutions. This new mode also provided the elements of the Kalman filter's covariance matrix.

MANS TIME INPUT

The MANS, as implemented on TAOS, needs an accurate time input to meet its output requirements. The specification on the spacecraft was that time provided to the payload would have an accuracy of 1.5 milliseconds. To achieve this accuracy, time is delivered from the CGU to the payloads and from the ground to the CGU. Analysis of all the sources of time error indicates that the total error of the MANS time input is on the order of one millisecond.

The Payload Executive sends time to the payloads every three seconds. Alternate 1553B channels are used so that if one channel fails entirely, the time will still be delivered every six seconds on the other channel. The six seconds will meet the seven and one half second time update requirement of the Dual Cone Scanners. The source of this time delivery is the digital hardware in the Payload Executive GVSC, which was not designed to have long term accuracy.

To solve this problem, the time in this GVSC is updated every minute by the CGU. The oscillator in the CGU is temperature compensated and provides a much more accurate time than the GVSC. The oscillator frequency can be altered by ground command in order to reduce the CGU drift rate.

To update the GVSC clock the Payload Executive dynamically passes bus control from the GVSC to the CDH, the main spacecraft computer. The CDH is interrupted by the CGU at the next 100 microsecond tick. Then the CDH loads the time into the 1553B transmit buffer to be sent to the GVSC. The value of this time includes a bias to take into account the time from the occurrence of the tick until the time can be sent out. Then the CDH returns bus control to the Payload Executive. It is generally not recommended to design dynamic bus control into a 1553B system, however this hand-off has operated during every minute of bus operation and has never failed.

It is also possible to determine the error in the CGU and uplink a command containing the amount of time to be added algebraically to the CGU time to remove this error. The CGU error is determined by reading the CGU time 59 times one second apart. The telemetry containing the 59 CGU time values is time tagged at the receiving Remote Tracking Site (RTS). The time tagging is achieved by multiplexing an IRIG B signal with the one megabit per second payload data at the RTS and transmitting the composite stream to the Consolidated Space Test Center (CSTC, more properly called Space and Missile Systems Center Detachment 2). The multiplexing eliminates the effect of delays occurring during the transmission of the composite signal from the RTS until it is demultiplexed at CSTC.

SENSOR SUBSYSTEM DESCRIPTION

The scanners and their electronics units are referred to as the MANS Sensor Subsystem (MSS). The MSS was built by the Barnes Engineering Division of EDO Corporation. Figure 3 is a photograph of the two scanners and their electronics units.

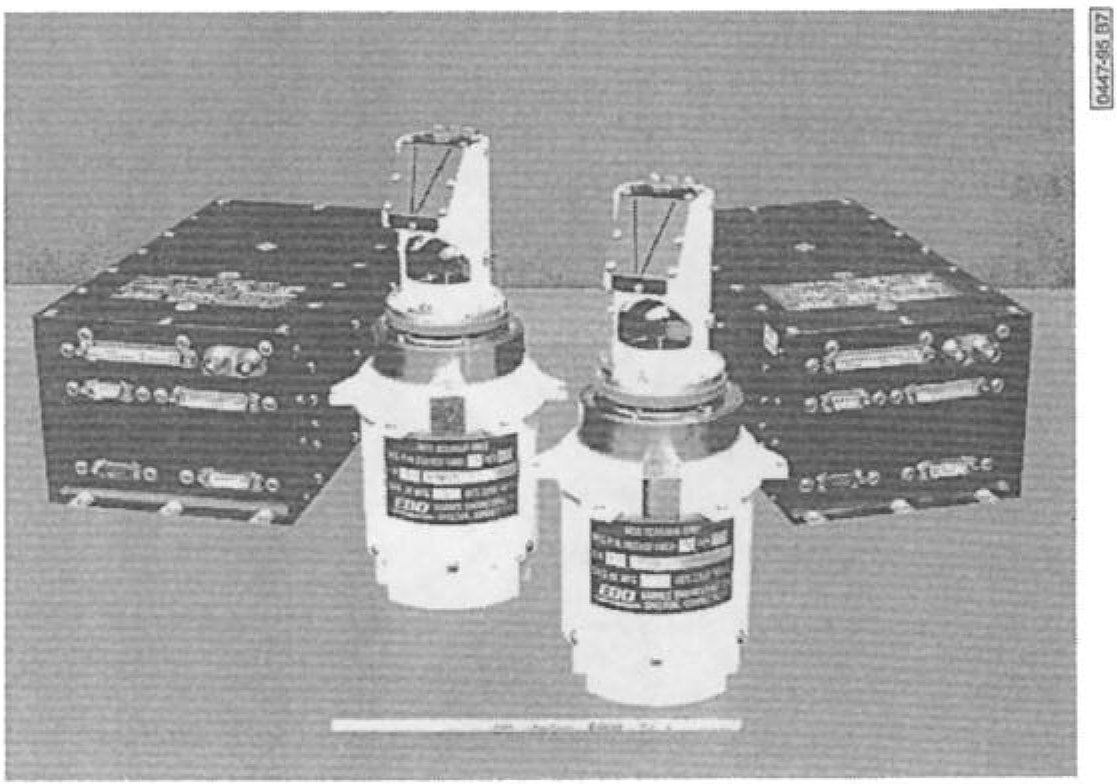

Figure 3. Photograph of the Dual Cone Scanners and their Electronic Units.

Each scanner weighs 3.4 pounds. The scanner and its electronics unit consume 0.5 amperes at 28 volts for a steady state power of 14 watts. To start up the scanner consumes 0.8 amperes for approximately 5 seconds. The scanners achieve their operating condition within 90 seconds of being turned on. The overall length of each scanner is 9.15 inches. Its mounting interface hole should have a diameter of 3.875 inches. The spinning part of the scanner possesses an uncompensated angular momentum of 0.04 foot-pound-second.

A single electronics unit weighs 8.8 pounds and is 11.2" by 5.1" by 4.3". The electronics unit uses the Barnes BX-1750 Space Processor board? based on the Bendix BX-1750 single chip microprocessor running at 20 MHz. This board has a processing capability of 1.5 MIPS. The infrared outputs of the electronics unit are the locations of the edges of the Earth and its radiance value. The Sun signal saturates the visible amplifier so the only Sun outputs are the edges of the Sun. The Moon signal does not saturate the visible amplifier so the Moon outputs are the Moon edges and the visible intensity values. The electronics unit provides the time when the scanner is at zero degrees.

Figure 4 depicts the DCS scan patterns. The moving part of the scanner rotates about the Scan Axis at 4 rotations per second. The visible light slits sweep out two fans in space. These slits admit light from 23 degrees from the Scan Axis to 87 degrees from the Scan Axis. Each slit is 2.5 degrees wide and 72 degrees long. Consider a longitudinal great circle on the observation sphere of the scanner. The slits are 2 degrees on each side of this great circle at their closest near the scanner’s equator. The arcs of the slits on the observation sphere form an angle of 16 degrees with the longitudinal great circle.

One of the infrared pencils is shown in Figure 4. It is 73 degrees from the scan axis and next to the left-most visible slit in the figure. The other infrared pencil is 38 degrees from the scan axis. It is 175 degrees away in azimuth from the one shown.

The geometry of the slits imposes limits on the visible coverage. Since the Sun and Moon are approximately 0.5 degrees wide, their images move into the slit and out of the slit. The pulse width of these objects is closer to the slit width than the angular size of the objects. The slit cannot detect all the way to 90 degrees from the Scan Axis, but only to 87 degrees. This is to provide clearance for thermal blankets or spacecraft protrusions and eliminates 5 percent of the sky near the plane orthogonal to the sensor spin axis.

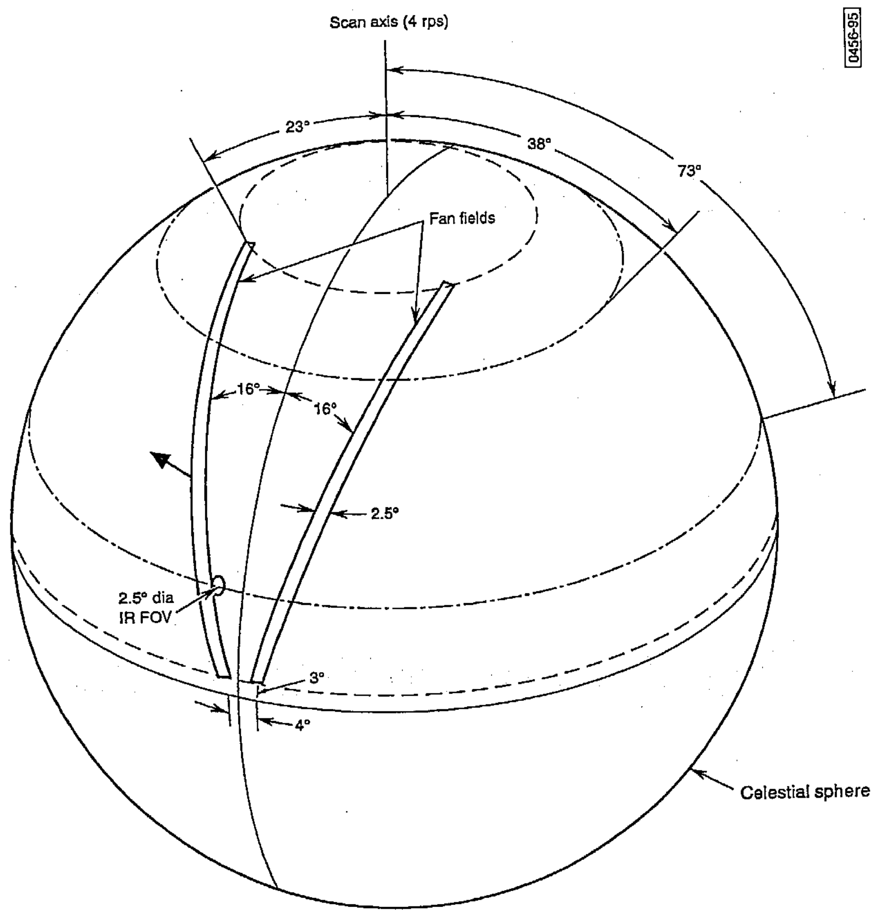

Figure 4. This is a sketch of the Dual Cone Scanner geometry. It shows the two visible slits and one of the infrared pencils. The other infrared pencil is 175 degrees away in azimuth and 38 degrees from the scan axis.

The greatest gap in the visible field of view is caused by the fact that the slit does not extend to the Scan Axis. Visible sources can only be seen if they are more than 23 degrees from the Scan Axis. This coverage hole is centered on the Scan Axis and covers 8 percent of the sky. This may or may not limit the MANS operation depending on where the scanners are placed on the spacecraft and how the attitude of the spacecraft is maintained.

SENSOR SUBSYSTEM ON-ORBIT OPERATION

There was an impediment to the performance of the MANS experiments on STEP Mission Zero because the spacecraft is nadir pointing and the attitude was controlled by yaw steering so that the solar arrays are pointed to the Sun to maximize power input. The nominal Sun viewing scanner is thus pointed such that its Scan Axis is in the plane defined by the spacecraft, the Earth center, and the Sun. At the satellite's altitude of 300 nautical miles the Earth horizon is approximately 67 degrees from nadir. Therefore at sunrise for the spacecraft, the Sun moves right into the blind spot. The Sun cannot be seen until it is above the blind spot at 113 degrees from nadir if the Sun happens to go that high. When the Sun is near the orbit pole, the nominal yaw steering algorithm would have kept the Sun in the sensor blind spot indefinitely. To get more Sun visibility, the spacecraft controllers yawed the satellite 30 degrees from the nominal sun-pointing direction during some of the MANS experiments. This reduced the power input by about 14 percent, which was tolerable.

On July 19, 1994, the STEP Mission Zero Inertial Measurement Unit failed. The absence of the EMU's gyroscopes meant that a new attitude control method had to be developed. Yaw steering to continually point the solar arrays toward the Sun became impossible. Ironically this usually meant that MANS observation of the Sun improved, but at the cost of reduced average solar power input.

The Moon channel was designed to have sufficient dynamic range so that it only saturates at twice the theoretical maximum full Moon intensity. The Moon signal processing determines the angular values of the leading and trailing Moon edges by adaptively adjusting the threshold at which the angles are read out. This threshold is set at one half the current maximum Moon intensity, which is determined by averaging 10 Moon amplitudes. This adaptive procedure provides accurate values of the Moon edges.

Experiments in the spring of 1994 indicated that there were occasions when the Moon pointing scanner was unexpectedly not able to observe the Moon. There was an initial concern that the scanner or its connecting cables were damaged during launch. Further investigation indicated that when the satellite was in sunlight the Moon signal in the visible scanner channel went to zero.

During tests in October, however, there were times when the Moon signal was evident before and during the satellite's eclipse period. The observations in the spring were made during the time that the angle between the orbit plane and the Earth-Sun line, the so-called beta angle, was small. When the satellite went from eclipse into sunlight it also went from being over a dark Earth to over a Sun-illuminated Earth. In the October test period the beta angle was approximately 60 degrees so that the satellite was in sunlight but was still over mostly dark Earth.

One hypothesis is that the reflected Earth light and not the direct Sun light that causes the Moon signal to go to zero. The cause of the problem could be the edge detection triggering software in the scanner's electronics unit, in which the Moon edge detection algorithm is incorrectly using the bright Earth signal when determining the maximum Moon signal. There is supposed to be a Moon pulse width assessment in the electronics unit's software to prevent this confusion, but this was never tested during the pre-launch scanner qualification testing. There was a Moon test with an interfering simulated Earth reflection but the amplitude of the Earth reflection was about the same amplitude as the Moon signal. When the amplitude of the reflected Earth is twice the Moon amplitude, the Moon signal could be lost entirely if the pulse width test does not operate correctly. The hypothesis that the pulse width test is not operating correctly appears consistent with the on-orbit results. This still needs to be verified by examination of the source code for the sensor electronics unit.

DATA VERIFICATION AND GENERAL PROCESSING

Most of the MANS flight code[8] is associated with on-board data correction, checking, and verification, rather than the actual orbit and attitude determination computations. The most important function of the autonomous flight code is the verification that the data being used are correct, both in terms of functioning sensors and software. Data identified as bad are flagged and not used in further processing and a notification is sent to the ground (or another spacecraft processor) that bad data have been encountered.

All of the data undergoes a variety of internal checks and checks against other sensor data as well. The nature of the checks depends on the sensors used. As described above, the TAOS configuration uses two Barnes Engineering Dual Cone Scanners for Earth horizon detection. These sensors provide 8 Earth horizon crossings every 250 milliseconds. First, the Sun on the horizon is identified and flagged. The remaining data are still well over-determined since only 3 points are needed to define a circle (i.e., the Earth's horizon) on the celestial sphere. These are used to identify any bad horizon crossing data points. If none are found, then the system has either 7 or 8 horizon crossings distributed around the disk of the Earth and the identification of the Earth can be regarded as extremely reliable. Similar data checks are used for other sensed objects or different sensor types.

The MANS software uses the patented Gontin-Ward[9] radiance profile correction method to reduce the variance in the height of the atmosphere. This method is believed to be able to reduce the angular error of the edge of the Earth to approximately 0.015 degrees.

In addition to internal checks, multiple sensor cross checks are used to further verify performance. Thus, the observed angular separation between the Sun and Moon is used to determine that the Moon is being seen and the angular separation between the Moon and the center of the Earth is used to verify that it is not "Los Angeles at night" that is being seen.

The validated sensor data are used to determine the attitude, update the Kalman filter that is modeling the orbit, and for several deterministic algorithms[*] using. Two separate time constants. As implemented on TAOS, MANS provides the following outputs at 250 millisecond intervals:

Attitude and attitude rate
Position and velocity (or standard Keplerian orbit elements)
Ground lookpoint for up to three spacecraft-fixed sensors
Direction to the Sun in spacecraft coordinates (whether or not the Sun is visible to the sensor)

The deterministic solutions for the position serve three distinct functions. They are used to initialize the Kalman filter; they provide moderate accuracy solutions until the Kalman filter converges; and they serve as a watchdog on the filter. The watchdog

*'The MANS algorithms are covered by U.S. Patent No. 5,109,346, "Autonomous Spacecraft Navigation System," issued April 28, 1992, to J. R. Wertz. Implementation of the software was done under the direction of L. J. Hansen.

function is important since it is always possible that a sophisticated filter, no matter how well designed, can diverge. Thus, a continuous comparison is made between the filter and the less accurate but very robust deterministic solutions. If the solutions differ sufficiently, MANS returns the deterministic solution, sets a flag to indicate less accurate results, and resets the Kalman filter. This process, summarized in Figure 5, is used to achieve robustness in the autonomous system. When hardware or software problems or anomalistic conditions cause the high accuracy Kalman filter solution to become unavailable, MANS adapts to provide the most accurate available solution.

On December 4, 1994, the GVSC devoted to MANS failed. The investigation into the exact cause of this failure is still under way. Before the computer was shut down by a CDH reset it was running programs from RAM and it was known that there was an EEPROM problem. There were EEPROM errors in every 4 Kilobyte page of the entire EEPROM memory space. It was clear that once the computer was turned off it was very unlikely that it would start up again.

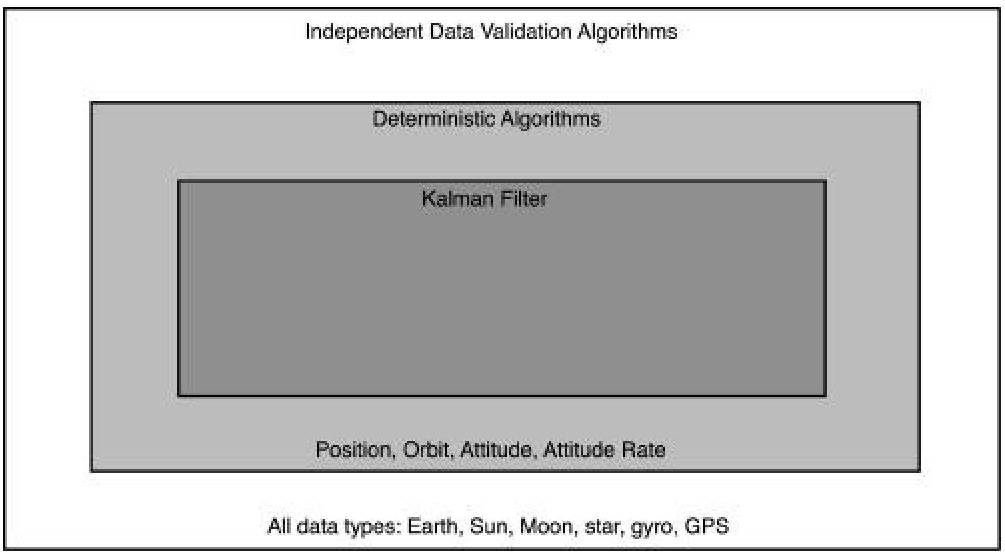

Figure 5. MANS provides robust autonomy by undertaking a series of tests and checks on sensor data. All data are validated before they are used for orbit or attitude determination. The best available data are used at any given time.

Fortunately, the contingency planning for the TAOS payloads anticipated the possibility of the failure of any experimental payload, including the Honeywell computers. The Dual Cone Scanner data can be entered as input to a GVSC flight computer in the laboratory. Thus the issue is the delivery of the sensor data to the experimenters on the ground. The Payload Executive can move the DCS data directly to the MMU. If the GVSC with the Payload Executive also fails, the CDH can move the DCS data to the MMU.

ON-ORBIT PERFORMANCE

Sensor problems with Moon detection, the computer failure, and the priority of other experiments that could only be conducted on orbit, have thus-far prevented a complete on-orbit performance evaluation of the MANS system. All of the flight code has successfully executed on orbit. The principal item that remains to be completed is the on-orbit calibration of the system in order to evaluate the accuracy which can ultimately be achieved. Fortunately, we believe that sufficient sensor data were obtained during an October experimentation period to allow calibration and tuning to be done on the ground and then to "fly" the system on the ground for direct comparison with ground-tracking data. This work is ongoing at the present time.

The on-orbit sensor data have now been well-characterized and exhibit noise levels at or below those needed to achieve the projected system accuracies. Noise levels on the Earth sensing are well below the required 0.1 degree level, even though the Gontin-Ward algorithm has not yet been applied. This is best illustrated by Figure 6 which shows the Earth-sensing noise levels (without Gontin-Ward) in terms of range to the center of the Earth. Noise in this figure is approximately ±150 meters in range, corresponding to approximately 0.08 degrees noise per horizon crossing.

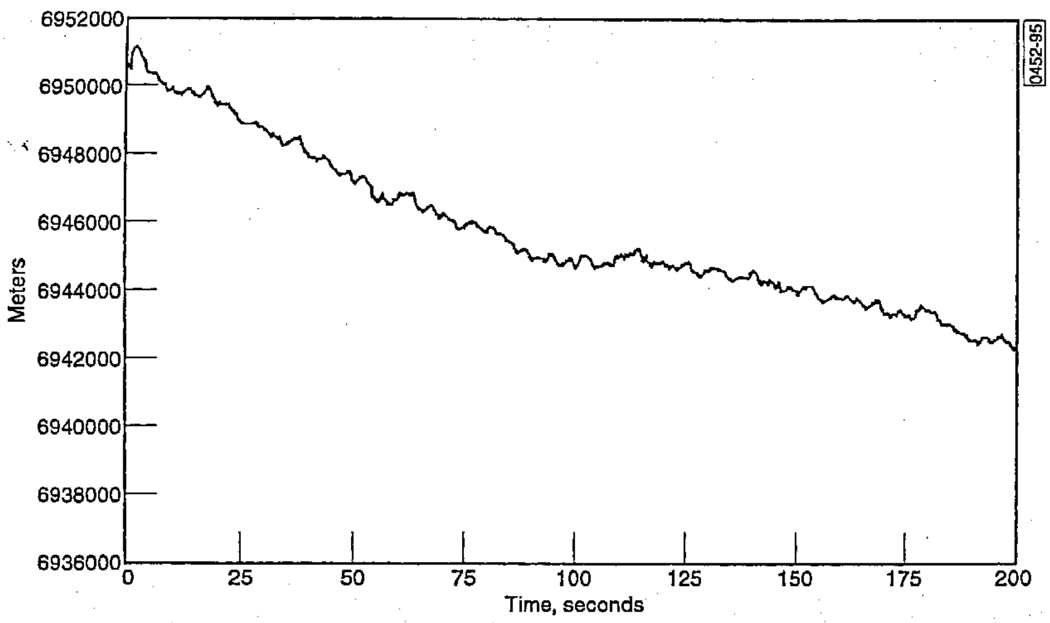

Figure 6. Noise levels on MANS measured range to the center of the Earth are at a level of approximately *150 meters. These values were determined on orbit over a 200 second time span.

Figure 7 shows individual measurements of the Sun azimuth angles in sensor coordinates in a typical, randomly selected, 6 second interval. The oscillation on the data is the attitude motion of the spacecraft* . The residual noise on the Sun data in this set of measurements is approximately ±0.005 degrees, which is substantially better than required.

Finally, Figure 8 shows the Moon azimuth angles in sensor coordinates. As expected, the Moon sensing noise is higher than that for the Sun and is a function of the Moon intensity. The intensity of a first or last quarter Moon is about 10% of the intensity

* On TAOS, MANS is running as an experiment and is entirely independent of the spacecraft control system.

level of the full Moon. The DCS is designed to sense the Moon down to a level of less than 4% of the intensity of the full Moon, but the Sun interference problems previously discussed have made this difficult to verify. The data for Figure 8 are for March 21, 1994, when the Moon was 9 days old and had an intensity in visible light of approximately 15% of the full Moon. The noise levels here of approximately ±0.02 deg are consistent with the requirements.

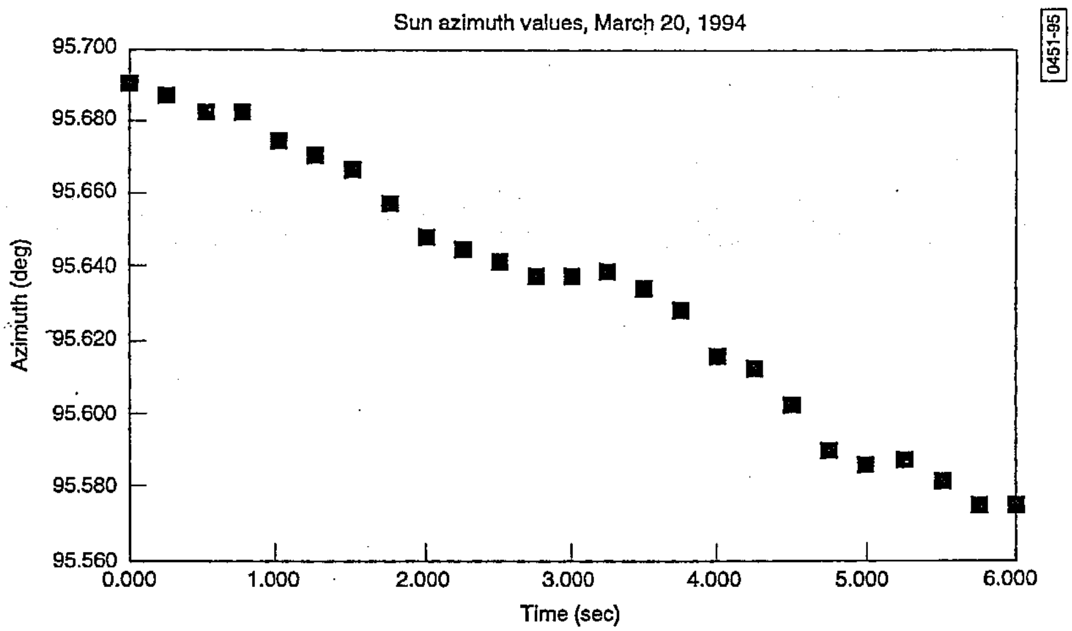

Figure 7. Noise levels on MANS measured azimuth angles for the Sun are approximately ±0.005 degrees.

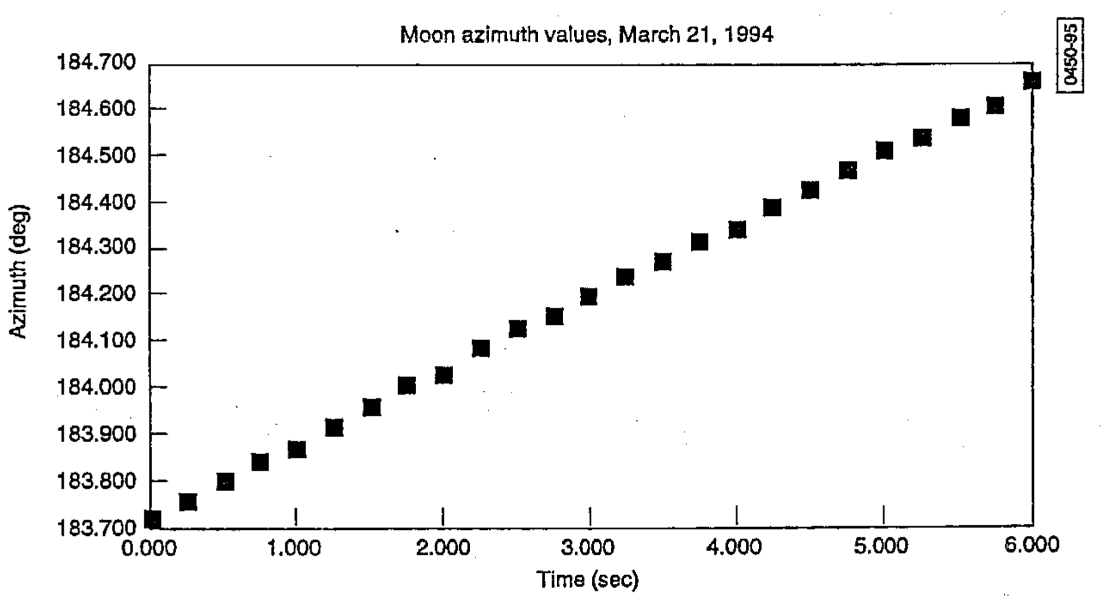

Figure 8. Noise levels on MANS measured azimuth angles for the Moon are approximately ±0.02 degrees for a 9 day old Moon (15 % of full Moon intensity) on March 21,1994.

CONCLUSIONS

At this time, data evaluation is still on going. The ultimate accuracy that can be achieved by MANS will depend on the reduction of systematic residuals by bias determination and tuning of system parameters. We anticipate that this can be done with the on-orbit data collected to date. The DCS sensor has difficulty detecting the Moon when the Sun is visible in the spacecraft sky. No problems have been seen in either Sun or Earth sensing. Noise levels on Sun, Moon, and Earth sensing are fully consistent with meeting the program goals. The work done over the next several months will reveal the ultimate accuracy of MANS.

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