Miniature Sensor Technology Integration (MSTI) -3 Pointing Error Analysis and Compensation

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Abstract. MSTI-3, an advanced technology demonstration satellite, was launched on May 16, 1996, to collect data in the short wave infrared (SWIR) and medium wave infrared (MWIR) bands. MSTI-3's mission was to survey the Earth: collecting data to support analysis of ground features, such as terrain and bodies of water; and atmospheric features, such as clouds and aurora. In April 1997, MSTI-3 analysts computed a pointing offset of approximately 50 miles. This paper summarizes the detailed fault analysis implemented by the operations team, including steps taken to isolate the cause of the pointing error to the payload mirror gimbal roll encoder and the spacecraft's GPS position board. Procedures implemented to compensate for the gimbal roll encoder and position error are also described, including how the mission control team was able to respond to the need for a precise pointing ability for a series of special operations.

MSTI Program

In December of 1991, the Ballistic Missile Defense Organization (BMDO) began the MSTI program with the goal of developing a means of providing rapid access to space to test new sensor technologies. Congressional direction at the end of FY94 transferred program management of the MSTI program from BMDO to the Air Force Space and Missile Center, SMC/MTAX. The final spacecraft in the program, MSTI-3, was designed to conduct a long duration, low-orbit measurement program to better understand and characterize clutter phenomena. The satellite's primary one year mission was to collect SWIR and MWIR background clutter data and scenes measuring temporal, nocturnal, and seasonal variations.

Due to a combination of changes in management and funding, and launch vehicle problems, the satellite development cycle extended longer than originally anticipated. MSTI-3 was finally launched on 17 May 1996 GMT from a Pegasus launch vehicle staged out of Vandenberg AFB. After the initial on-orbit checkout, the spacecraft was operated out of the MSTI Operations Center (BATCAVE) located in Alexandria, VA.

MSTI-3 Specifications

MSTI-3 basic spacecraft design included a fixed, three-faceted GaAs solar array attached to an aluminum structure built up around the propulsion system. Spacecraft electronics and subsystems were housed in either one of two electronic bays or affixed to the outside of the aluminum structure. The MSTI-3 spacecraft, benefiting from lessons learned during the two prior missions, included improvements in the attitude control, power, and data handling subsystems (see figure and table in next column).

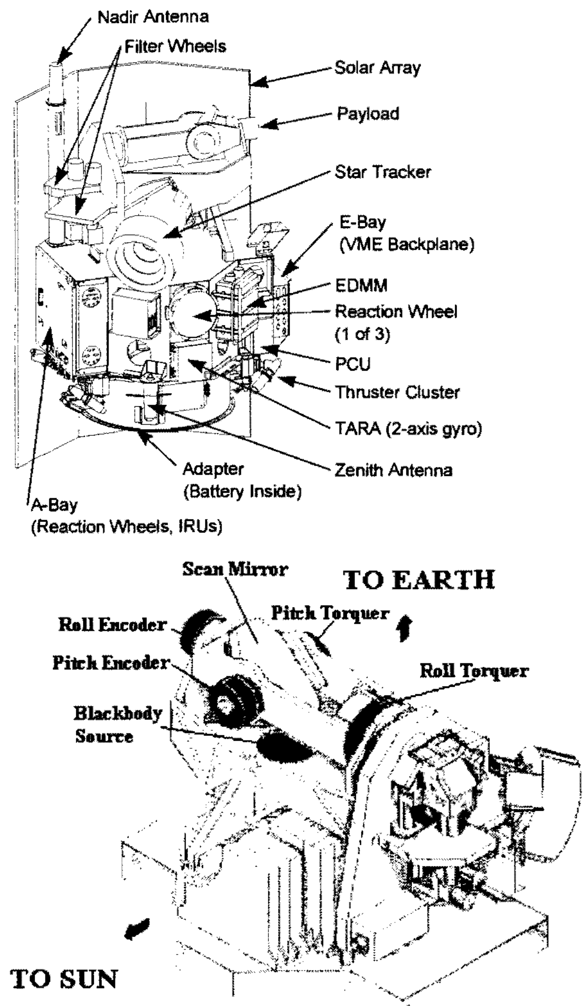


Figure 1. MSTI-3 Line Drawing and Payload Schematic

SPACECRAFT PARAMETER	CHARACTERISTIC
DESIGN LIFE	1 Year 85 % Mission Effectiveness (EOL)
SPACECRAFT DIMENSIONS	32" Diameter, 56" Height
SPACECRAFT MASS PAYLOAD MASS PROPELLANT MASS	466 LB (includes P/L and Prop) 115 LB 47 LB
PROPULSION	Hydrazine (for orbit adjust and reaction wheel desaturation)
ATTITUDE DETERMINATION AND CONTROL	3 - Axis Reaction Wheels GPS/Star Tracker
POWER	225 Watt Solar Array (EOL) 10 A-Hr NiH Battery Cruise/Observation - 140 W/320 W
STRUCTURE/THERMAL	Payload thermally isolated from bus
COMMAND AND DATA HANDLING	1750 A - Spacecraft RISC 3081 - Payload
DATA STORAGE (EDMM)	8.64 Gbits @ 25 Mbits/s
TT&C	SGLS - 2 kbit/s Commanding (with PRN range capability) - 32 kbit/s SOH - 1 Mbit/s Mission Data

Table 1. MSTI-3 System Specifications

Both the MSTI-3 bus and payload were designed to provide pointing control, low jitter, and a high field-of-regard. The spacecraft was three-axis stabilized and used reaction wheels to maintain its nominal pointing attitude. The spacecraft payload consisted of three cameras: a short-wave infrared (SWIR), a medium-wave infrared (MWIR), and a visible spectrometer. In addition, the spacecraft had a star tracker which was rigidly attached to its payload supports, replacing the horizon sensor flown on previous missions. A GPS system was also added to enable enhanced position and velocity determination.

June 1997 marked the end of MSTI-3's primary mission. At this point, over 1.2 million images with a minimum resolution of 40 meters had been collected. Although designed for one year, the satellite was deemed capable of producing additional data for an extended period of time. Based on cryo-cooler constraints, both infrared instruments were predicted to operate within specifications for at least a year past the primary end of mission date. The visible instrument, which had been rarely used due to the nature of the SBIRS science requirements, was expected to match or exceed the infrared instruments' life estimates. Similar to the previous year, however, it was noted that there would be significant operational limitations placed on the spacecraft during the upcoming 1997 eclipse season.

Several organizations expressed interest in the MSTI-3 spacecraft and requested that the Air Force investigate the possible extension of its mission. Subsequent funding was secured to support additional program operations through November 1997.

Before program funds were exhausted, SMC/MT directed ANSER to convene a tiger team consisting of SMC/TEO, ANSER, The Aerospace Corporation, and USSPACECOM to formulate a de-orbit plan. NASA was later brought on board when collision avoidance issues concerning both the *Mir* and *Columbia* arose. Final authorization to proceed with the satellite de-orbit plan was received on 01 December 1997 and the MSTI-3 spacecraft was successfully completed a controlled reentry on 11 December 1997.

MSTI-3 Pointing Error

The nature of the spacecraft's primary mission translated into a target set of clouds and other upper atmospheric features. Due to this fact, routine correlation of imagery to ground truth was not possible. To compensate, ground truth pointing verifications were performed on a semi-regular basis. Several months prior to the end of MSTI-3's primary mission, image analysts noticed that for one of these ground truth verification experiments, image frames collected by the payload did not display any of the expected geographic features. An image mosaic of the collected data was created and the land structures exhibited in the image data were compared with regional maps. After the imaged location was identified, the team computed a pointing error of approximately 50 miles.

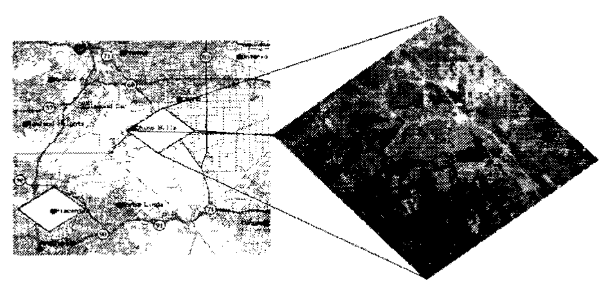


Figure 2. Map showing initial pointing error

The experiment was planned to collect images of the Anaheim, CA area. As depicted in the processed image and corresponding map above, however, the payload cameras actually recorded images near Chino Hills, CA—nearly 50 miles to the northeast.

At this point, two independent investigations to isolate contributing factors to the pointing error commenced. The first investigation utilized a more traditional approach wherein a fault-tree analysis of the pointing error was created*. Although the fault-tree served to identify which systems contributed to on-board pointing calculations, the analysis was not able to single out a likely cause of error.

After each payload operation, spacecraft analysts compared reported pointing parameters received in the spacecraft telemetry to planned pointing parameters. Although ground truth suggested that the spacecraft was not pointing correctly, spacecraft telemetry indicated that all systems were functioning nominally. Thus, the second approach attempted to duplicate and verify each of the reported telemetry parameters that contributed to the calculation of an accurate pointing (or line-of-sight) vector.

Calculation of Line-of-Sight (LOS) Vector

The scanning mirror assembly contained within the spacecraft's payload reflects the visible and infrared electromagnetic radiation from a target or background into the direction of the fixed telescope. The assembly consists of a flat aluminum mirror mounted on a gimbal with two degrees of freedom. As illustrated in Figure 3, The mirror can roll about the spacecraft x-axis (referred to as ϕ), and pitch about the spacecraft y-axis (referred to as θ). The motion of the two axes is provided by two DC motors (torquers). The angular position of each axis is determined by 18 bit absolute encoders located on the opposite side of each torquer. A digital controller receives the information containing the pointing instructions and compares it to the position data from the encoders. The controller positions the motors based on the results of the comparison and the selected control scheme[i].

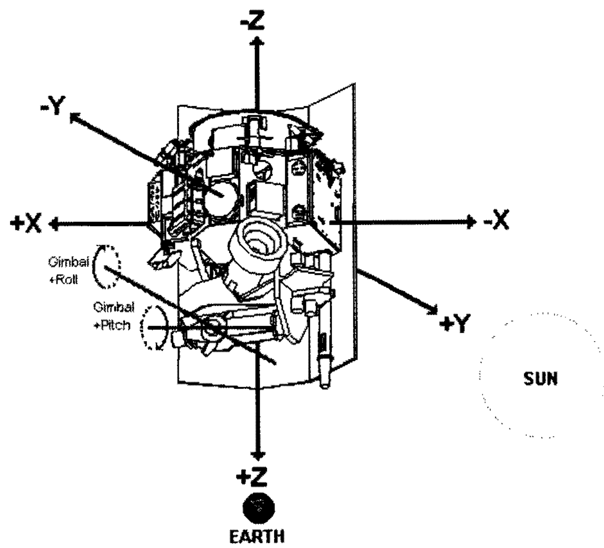


Figure 3. Spacecraft and Payload Gimbal Coordinate Definitions

The light path of the LOS boresight includes two reflections: one from a fixed 90° beam director and the other from the aforementioned payload scanning mirror assembly.

The LOS vector can be calculated as a unit vector in spacecraft platform coordinates by the following equation[ii]:

$$LOS_{platform} = \begin{bmatrix} -\cos(2\theta) \\ -\sin(2\theta)\sin(\phi) \\ \sin(2\theta)\cos(\phi) \end{bmatrix}$$

where the roll (ϕ) and pitch (θ) values are provided by the payload mirror gimbal encoders.

* Fault-tree analysis is included as Attachment 1

Using the spacecraft quaternion provided by the star tracker, the LOS vector can then be expressed as a unit vector in the J2000 Earth Mean Equator inertial reference system which is centered at the earth's center. This reference system is also referred to as Earth Center Inertial (ECI). The four quaternion elements provided by the star tracker (q_1 through q_4) can be used to express the rotation, Q, from spacecraft platform coordinates to the inertial reference system:

$$Q_{platform}^{ECI} = \begin{bmatrix} q_1^2 - q_2^2 - q_3^2 + q_4^2 & 2(q_1q_2 - q_3q_4) & 2(q_1q_3 + q_2q_4) \\ 2(q_1q_2 + q_3q_4) & -q_1^2 + q_2^2 - q_3^2 - q_4^2 & 2(q_2q_3 - q_1q_4) \\ 2(q_1q_3 - q_2q_4) & 2(q_2q_3 + q_1q_4) & -q_1^2 - q_2^2 + q_3^2 + q_4^2 \end{bmatrix}$$

Once the LOS vector has been expressed in the inertial reference (ECI) system, the vector can be rotated into the Earth Center Fixed (ECF) coordinate system. This system differs from the inertial reference system by only a rotation about the celestial z-axis. This rotation is known as the Greenwich Hour Angle (GHA), and is referred to by the symbol λ.

The rotation matrix needed to transform the LOS vector from the ECI coordinate system to the ECF coordinate system is as follows:

$$R_{ECI}^{ECF} = \begin{bmatrix} \cos(\lambda) & \sin(\lambda) & 0 \\ -\sin(\lambda) & \cos(\lambda) & 0 \\ 0 & 0 & 1 \end{bmatrix}$$

In summary, through matrix multiplication, the LOS vector can be expressed as the combination of the three previous matrices as follows:

$$LOS_{ECF} = R_{ECI}^{ECF} Q_{platform}^{ECI} LOS_{platform}$$

Error Analysis Using Calculation of LOS Vector

To isolate and individually analyze potential error sources within the LOS calculation, contributing parameters to each of the three matrices which combine to produce the LOS vector in ECF coordinates were investigated in turn.

First, the roll and pitch values of the spacecraft contribute to the calculation of the initial LOS vector which is expressed in spacecraft platform coordinates. As mentioned previously, these values are reported by the payload mirror gimbal encoders. Secondly, the matrix used to rotate from spacecraft platform coordinates to ECI coordinates relies on the attitude quaternion provided by the star tracker. And finally, the matrix needed to rotate the LOS vector from ECI coordinates to ECF coordinates depends only on the GHA. This angle is calculated based on the time reported by the spacecraft's GPS receiver. Thus, by calculating and verifying each step in the LOS calculation, the values of roll, pitch, attitude quaternion, and spacecraft time could be isolated and/or accepted or rejected as contributors to the pointing error.

Calculation of LOS Vector in Platform Coordinate System

The first test utilized was designed to isolate the roll and pitch values as potential sources of error introduced in the initial calculation of the LOS vector in platform coordinates. The experiment was able to isolate these values by creating a map of the thermally calibrated blackbody which was located on the underside of the spacecraft's payload. The charts shown on the next page are contour maps created from three-dimensional depictions of the warm calibration plate where the x- and y-axis represented roll and pitch, and the z-axis represented the average pixel return at each roll and pitch location. In order to identify at what point the actual maximum return occurred, sample images were collected in 2 degree intervals for both roll and pitch. When the spacecraft was operating nominally, the maximum return from the calibration plate occurred at a roll value of 180 degrees and a pitch value of 45 degrees, as shown below in the chart below.

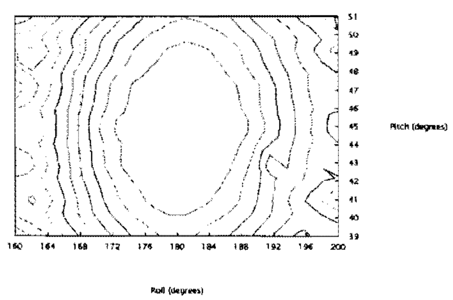


Chart 1. MWIR Warm Calibration Plate Mapping Without Error

After the discovery of the pointing error, the maximum return was received at a roll of 196° and a pitch of 45° as shown below.

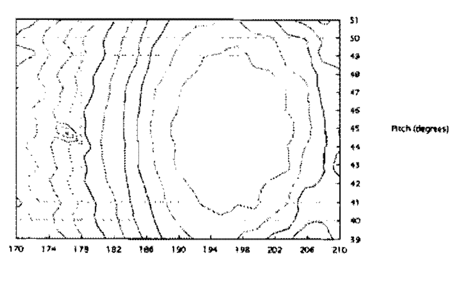


Chart 2. MWIR Warm Calibration Plate Mapping with Roll Error

Due to the fact that the roll value had changed significantly for the location of maximum return while the pitch value had remained constant, this experiment served to indicate that the reported roll value was a source of the pointing error. The payload mirror roll position is reported via an 18-bit roll encoder. The equation to convert from raw roll counts to degrees is below[iii]:

$$\phi = \left(Roll_{counts} + Roll_{bias} - 73000\right) * 1.373291e^{-3}$$

Prior to launch, the roll bias shown in the previous equation was calculated so that a roll value of 0° would align the LOS vector with the spacecraft's z-axis[iv]. Since the reported roll value now deviated from its known value, the roll bias was modified to compensate for the 16° shift.

The warm calibration plate mappings were repeated twice each week to monitor the roll value. Due to the fact that these experiments only provided up to 2 degrees of accuracy, this process was not used for fine determination of the roll bias. The warm calibration plate mappings, however, did serve as an on-orbit means of initially isolating and measuring the gross adjustments of the roll bias that were required to minimize the pointing error.

Platform to ECI Rotation Matrix

Once the reported roll and pitch values were verified and/or compensated, parameters which feed into the platform to ECI rotation matrix were analyzed. The rotation matrix that is used to express the LOS vector in the inertial reference system relies only on the attitude quaternion provided by the on-board star tracker. The star tracker functions by identifying stars within the image provided by the star tracker's charge-coupled device (CCD) camera. The angles between these stars are used in conjunction with an on-board star catalog to uniquely identify the LOS vector of the star tracker camera boresight and create the spacecraft quaternion. On at least one occasion, the star tracker provided an invalid solution due to incorrect identification of one star in the star tracker's FOV. The star that was incorrectly identified was similar in magnitude and angular position to another entry in the star catalog. To prohibit a similar occurrence, tolerance of the selection criteria was decreased and easily misidentified stars were removed from the star catalog.

Although the star tracker provided incorrect attitude quaternion on occasion, its solution was nominally reliable. Using this information, a second experiment was developed to verify the quality of the roll and pitch values. The satellite was commanded to image selective stars which are particularly responsive at infrared wavelengths. During these experiments, the spacecraft payload performed an initial inertial-fixed stare (IFS) at a particular location in space (using right-ascension and declination coordinates). After several seconds, the payload then performed a series of relative re-points of the focal plane in the ±x and/or ±y direction. The intent of the experiment was to align the LOS vector with an imaginary vector from the satellite to the star. If the reported roll and pitch values were correct, this experiment would locate the star in the focal plane as shown in Figure 4.

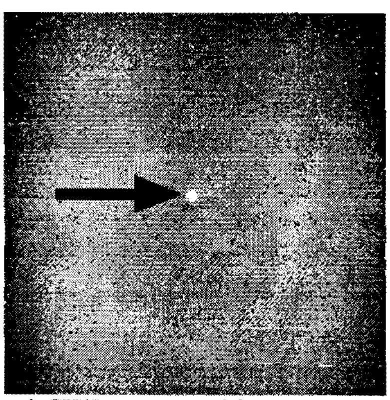


Figure 4. SWIR Image of Star without Error

In Figure 5 below, however, the star still appeared off center by some small angle due to the fact that the warm calibration plate experiment was only able to produce coarse roll error estimates.

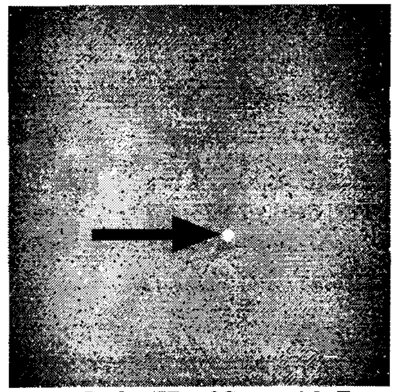


Figure 5. SWIR of Star with Error

Using the knowledge of the angular separation of each pixel and the orientation of the satellite, a new roll bias could be calculated. This method proved to be more exact than that of calibration source mapping and allowed reported roll error estimation to a precision of 0.016°. Celestial observations were repeated twice a week, but on alternate days than that of the calibration source mapping. This approach allowed for daily monitoring of the reported roll error, while also maintaining the integrity of two independent evaluations of the reported roll and pitch values.

ECI to ECF Rotation Matrix

The final parameter needed for the LOS vector calculation is the Greenwich Hour Angle (GHA). This parameter is defined by computing the angle between a vector from the center of the Earth to the point of 0° latitude and 0° longitude in the equatorial plane (which is constant) and a vector from the center of the Earth to the first point of Aries (which varies with time of day). This value is easily computed when the spacecraft time is known accurately. The on-board computer performed this calculation using the time provided by the GPS receiver. Although other problems were discovered with the GPS receiver, the reported time was assumed to be reasonably accurate.

LOS Vector-Earth Intercept Point Error

Once the LOS vector was transformed to the ECF coordinate system, the intercept point of the LOS vector on the earth's surface was required to determine the geographic location of the observed area. Calculation of this vector was relatively straightforward. To compute the earth intercept point, only one additional vector was required—the ECF position of the satellite, which was provided by the spacecraft's GPS receiver or the on-board propagator if the GPS was disabled. As illustrated in Figure 6 below, after determining these two vectors, the earth intercept point is simply the LOS vector subtracted from the spacecraft position vector.

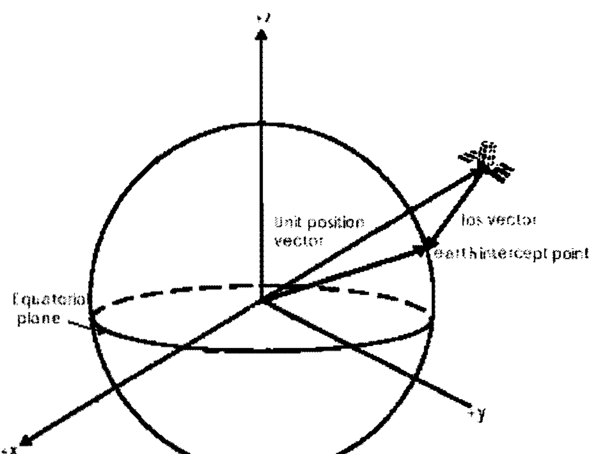


Figure 6. Earth Intercept Point

GPS Receiver

As was discovered early on in the mission, the spacecraft's GPS receiver did not reliably or accurately report the position of the spacecraft. This anomaly was reinforced during the acquisition of imagery over well-known coastal areas. These images, when compared with detailed geographic maps, provided a source of truth regarding the actual location of the image data. GPS position errors exceeding 50 miles were documented when the actual location of an image was compared to the computed location of the earth intercept point using the spacecraft position provided by the GPS receiver. Using ground propagated two-line element sets, rather than the GPS provided position, the calculated earth intercept point was much more reasonable, often within a half of a field-of-view (FOV). Due to the fact that other spacecraft functions depended on the accurate knowledge of spacecraft position, the GPS receiver was disabled in favor of uploaded state vectors which were fed to an on-board propagator.

With this method, the position error of the spacecraft degraded with time. The daily drift associated with the on-board propagator was expected to be between 3.4 km and 7.0 km per 6 hours[v]. To minimize this drift, state vectors were uploaded to the spacecraft over evenly space intervals approximately four times a day. As illustrated with the following chart, each vertical drop represents the update of the spacecraft position via the use of the ground provided state vector.

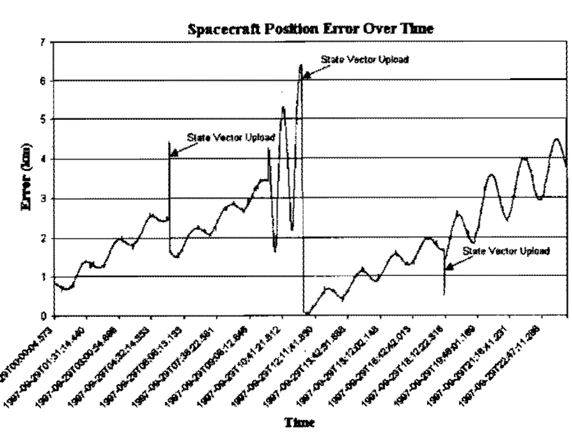


Chart 3. Orbital Position Error with Time

Once the spacecraft's reported position error was stabilized, acquisition of ground image data in well-known areas became the third monitoring procedure adopted to maintain the roll bias. In a similar process to that of star observations, images of geographically recognizable areas were acquired. Again, each experiment was planned so as to place the geographical region of interest in the center of the focal plane. As shown in the image below, this did not always happen. Using the same technique developed for the star experiments, the angular separation from the boresight was used to determine the roll bias value.

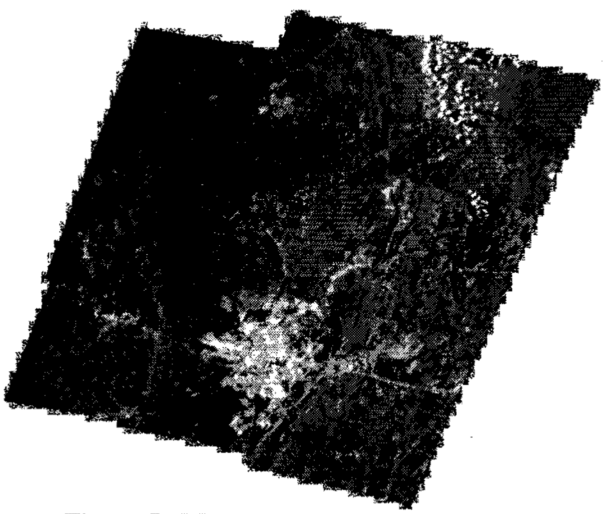


Figure 7. MWIR Image of Frederick, MD with Error

Roll Bias Unpredictability

Once isolated as a contributor to the pointing error, the reported roll bias required constant monitoring and modification. As shown in the chart in the next column, after the first few roll bias calculations and uploads, the gimbal mirror roll encoder began to float unpredictably and daily pointing tests—either warm calibration plate mappings, ground truth experiments, and/or star experiments—were required to monitor each variation. When a large error was identified in the roll bias error calculation, we uploaded a bias value to correct the error. Using these measures, the pointing error was minimized to the least degree possible.

Operationally, the unpredictable variation of the roll bias translated into a day-to-day uncertainty of the spacecraft's pointing error. For most experiments, the roll bias uploads adequately compensated for most of the pointing error. For other experiments, the roll bias upload was not sufficient in and of itself and necessitated the development of additional measures. These measures, developed and tested while on-orbit following the end of MSTI-3's primary mission, essentially allowed spacecraft operators to steer and hold the payload camera's FOV over specific geographic areas of interest in real-time.

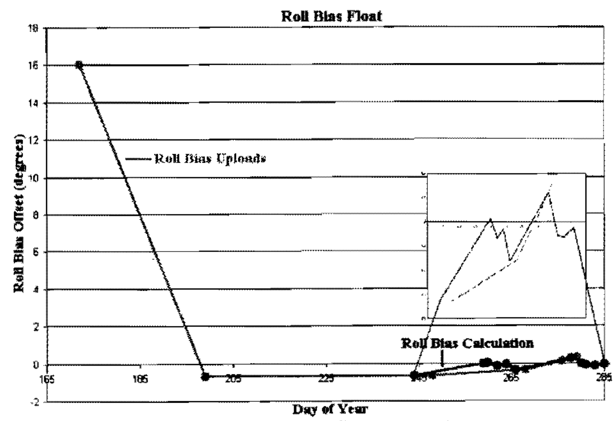


Chart 4. Roll Bias Float over Time

Before spacecraft operators could distinguish geographic features used to steer the FOV, image data from the payload had to be fully processed, calibrated, and displayed. The image processing team developed a statistical pseudo-calibration algorithm to approximate calibration parameters and create near real-time images from the payload data which was flowing from the spacecraft at the rate of 1 MBPS. Due to processing limitations, the time lapse between when an image was taken and when the calibrated image appeared on the ground was approximately 3-5 seconds. This time lapse, however, did not prove to have significant impact on the spacecraft operator's ability to steer the FOV.

Since the real-time steering of the MSTI-3 payload was not the nominal mode of operation for this system, new command scripts had to be designed and tested as well. A steer script is made up of roughly 8 - 10 commands. The payload design necessitates temporarily disabling payload data output when configuring the system to input data. Therefore, the first three commands in the sequence configure the payload to receive commands and *temporarily* disable payload data output. The next 5 commands execute the re-point and the last commands re-enable the payload data output. This whole sequence of 8 - 10 commands takes less than a second to execute when command capability of the spacecraft is nominal. Since real time downlink bandwidth requirements imposed a one Hertz image sample rate, one second payload output interruption would cause at the most 1 or 2 lost image frames.

Conclusion

Although the pointing error experienced by the MSTI-3 spacecraft proved to be a major impediment to mission operations, significant lessons can be drawn from the MSTI-3 team's experience. The information collected beyond the spacecraft's primary one year mission proved to represent a unique and important data set that would have been missed had measures to overcome the pointing error not been developed. Most spacecraft on-orbit represent the culmination of years of effort and millions of dollars of expense. Due to this fact, every effort should be made to exploit these systems to the fullest extent possible—often through the development of new operational concepts and experiments not accounted for in the original spacecraft design. The MSTI-3 program can certainly be characterized as a success in this regard.

References

i. *MSTI-3 Payload Software User's Manual Version 5.0.* SAIC. 26 April 1996.

ii. *Line-of-Sight (LOS) and Relative Repoint Calculation.* Ian Mitchell. MEM3A-VG-000022, 05 October 1995.

iii. [SAIC96]

iv. *Spacecraft and Payload Alignment Performed July 1996.* Ian Mitchell. MEM3A-IS-400021, 30 August 1996.

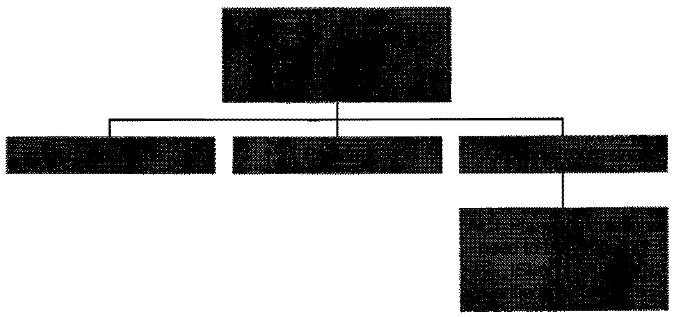
v. *On-Board Orbit Propagator Characterization.* Rob McEwen. MEM3-ISI-400061. 25 July 1997.

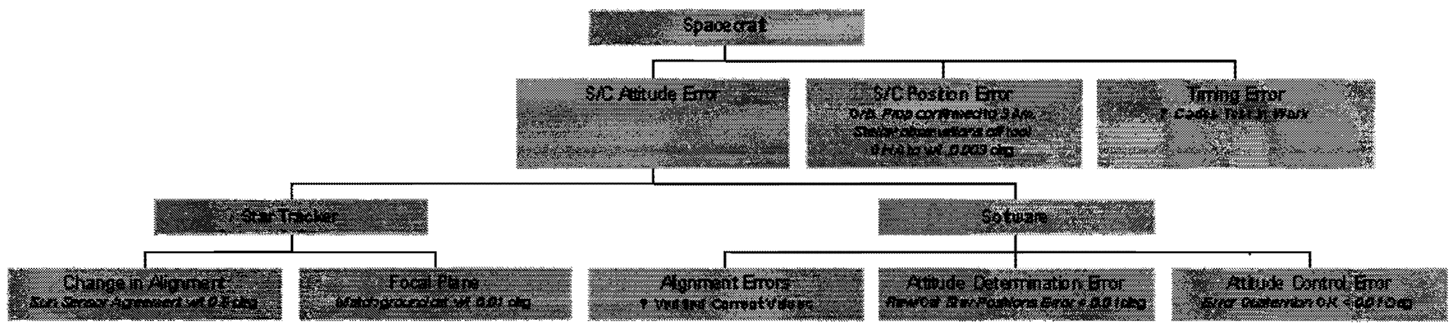
Biographies

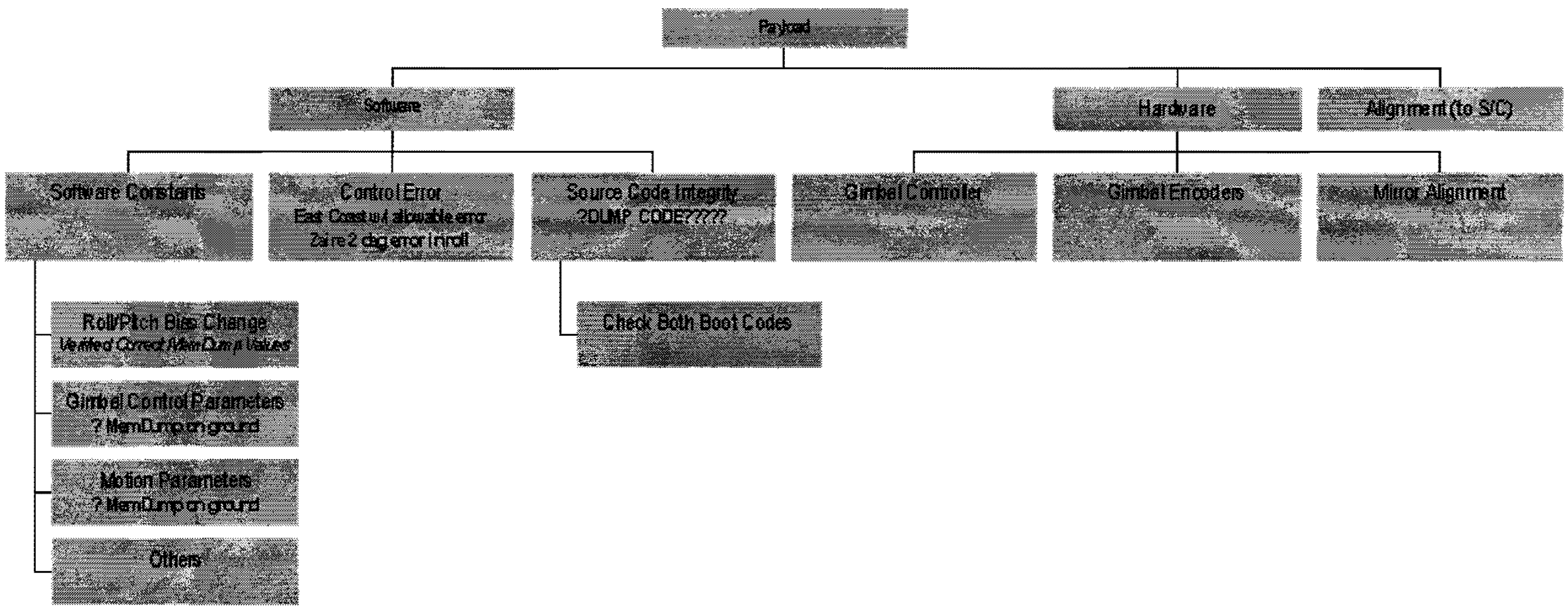
Howard W. Taylor has a Bachelor's degree in Electrical Engineering from the University of Delaware and has just completed requirements for a Master's degree in Electrical Engineering—also from the University of Delaware. Mr. Taylor joined Applied Coherent Technology Corporation, located in Herndon, VA, in 1994, where he performed image processing tasks associated with DSPSE/Clementine satellite program. At this time, Mr. Taylor also began his work with pointing geometry calculations. From 1995 to 1998, Mr. Taylor worked on the MSTI-3 satellite program in the area of image processing. Currently, Mr. Taylor is supporting the Stardust satellite program, as well as the Naval EarthMap Observer (NEMO) satellite.

Lesley M. Rahman has a Bachelor's degree in Mathematics from Carroll College in Helena, MT and a Master's degree in Space Studies from the University of North Dakota. Ms. Rahman joined Analytical Services, Inc. (ANSER) in 1997, where she performed mission planning tasks for the MSTI-3 satellite program. In 1998, Ms. Rahman began employment at AlliedSignal Technical Services Corporation (ATSC) as a member of the mission operations team for the newly formed DataLynx™ program. The DataLynx™ program is an ATSC commercial offering to provide complete end-to-end ground segment services for multiple satellite customers. The Naval EarthMap Observer (NEMO) satellite is DataLynx™'s initial customer.

Attachment 1 - Fault-Tree Analysis




Spacecraft
S/C Attitude Error
S/C Position Error
Timing Error
Star Tracker
Software
Change in Alignment
Focal Plane
Alignment Errors
Attitude Determination Error
Attitude Control Error


Payload
Software
Hardware
Alignment (to S/C)
Software Constants
Control Error
East Coast w/ allowable error
Zaire 2 deg error in roll
Source Code Integrity
?DUMP_CODE?????
Gimbal Controller
Gimbal Encoders
Mirror Alignment
Roll/Pitch Bias Change
Verified Correct MemDump Values
Check Both Boot Codes
Gimbal Control Parameters
? MemDump on ground
Motion Parameters
? MemDump on ground
Others