Attitude Accuracy Study for the Earth Observing System (EOS) AM-1 Spacecraft*

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ABSTRACT

Earth Observing System (EOS) spacecraft will make measurements of the earth's clouds, oceans, atmosphere, land and radiation balance. These EOS spacecraft are part of the National Aeronautics and Space Administration (NASA) Mission to Planet Earth, and consists of several series of satellites, with each series specializing in a particular class of observations. This paper focuses on the EOS AM-1 spacecraft, which is the first of three satellites constituting the EOS AM series (morning equatorial crossing) and the initial spacecraft of the EOS program. EOS AM-1 has a stringent onboard attitude knowledge requirement, of 36/41/44 arc seconds (3σ) in yaw/roll/pitch, respectively.

During normal mission operations, attitude is determined onboard using an extended Kalman sequential filter via measurements from two charge-coupled device (CCD) star trackers, one Fine Sun Sensor, and an Inertial Rate Unit. The Attitude Determination Error Analysis System (ADEAS) was used to model the spacecraft and mission profile, and in a worst-case scenario with only one star tracker in operation, the attitude uncertainty was 9.7/11.5/12.2 arc seconds (3σ) in yaw/roll/pitch. The quoted result assumed the spacecraft was in nominal attitude, using only the 1-rotation per orbit (rpo) motion of the spacecraft about the pitch axis for calibration of the gyro biases. Deviations from the nominal attitude would show greater attitude uncertainties, unless calibration maneuvers which roll and/or yaw the spacecraft have been performed; this permits computation of the gyro misalignments, and the attitude knowledge requirement would remain satisfied.

INTRODUCTION

Purpose and Methodology

Attitude error analysis studies are needed to determine whether the EOS-AM1 satellite can meet its attitude accuracy requirements using its onboard computer and sensor complement.

The present study determines:

1. The accuracy to which attitude sensor and gyro calibrations can be performed.

2. The expected attitude determination error for various sensor combinations.

The Attitude Determination Error Analysis System (ADEAS) is an analysis software tool that provides a general-purpose linear error analysis capability for various spacecraft attitude geometries, sensor complements, and determination processes. An appropriate NAMELIST setup permits ADEAS to model the salient features of the EOS AM-1 spacecraft and mission profile.

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Background

EOS-AM1 is the first of three satellites constituting the EOS-AM (morning equatorial crossing) series in support of NASA's "Mission to Planet Earth" and is the initial spacecraft of the EOS program.

EOS-AM1 will be launched from the Western Test Range, Vandenberg Air Force Base, California, with the General Dynamics Atlas IIAS launch vehicle in June 1998. EOS-AM1 has a planned mission lifetime of 5 years.

The initial parking orbit has an altitude of 525 by 705 km with an inclination of 98.2 deg. After a series of orbit-raising maneuvers, the mission orbit will be described as follows:

- Sun-synchronous polar
- Inclination = 98.2 deg
- 10:30 a.m. ±15 min descending node, local mean solar time
- 705 km altitude, circular
- Groundtrack repeats in 233 orbits/16 days, with ±20 km cross-track error at node crossings

Spacecraft Attitude Control System

The EOS-AM1 spacecraft is being manufactured by Lockheed-Martin, Valley Forge, Pennsylvania. The spacecraft will have the following complement of attitude sensor and actuator hardware:

- CCD star tracker (CCDST) (2)
- Earth scanner assembly (ESA) (2)
- Fine sun sensor (FSS) (1)
- Three axis magnetometer (TAM) (2)
- Coarse Sun sensor (CSS) (9 pairs)
- Inertial rate unit (IRU) (6 axes)
- Reaction wheels (4)
- Magnetic torquer rods (3)

Immediately after launch, the onboard computer, via the attitude thrusters, uses ESA data to roll and pitch the spacecraft to acquire the Earth and to orient the spacecraft body Z axis to nadir-pointing, then performs orbital gyrocompassing to align the body X axis roughly parallel to the velocity vector. When in normal mission mode, attitude is determined via sequential filter using the two CCDSTs (the FSS can substitute for one CCDST if one fails) and the IRU and is controlled by the reaction wheels.

The onboard computer uses an extended Kalman sequential filter to determine attitude during normal mission mode operations. The state vector is composed of an attitude quaternion and the IRU rate biases. Star observations are taken from alternate star trackers, and a particular star is used as a valid observation only if (1) it is in the onboard star catalog, and (2) it is visible in two consecutive observations by the same star tracker (16.384 sec later). The attitude propagation cycle time is 0.512 sec, based on using filtered gyro rates (the measured gyro data are available every 0.128 sec).

Attitude Requirements

The spacecraft attitude is described by a 3-1-2 (yaw-roll-pitch) Euler rotation sequence, which relates the body coordinate system (BCS) to the orbital coordinate system (OCS). The spacecraft null attitude has the BCS coincide with the OCS. The OCS is a rotating coordinate system. The OCS coordinate axes originate in the spacecraft's center of mass. The +Z axis points to the geocenter, the +Y axis points to the negative orbit normal, and the +X axis completes the orthogonal triad. The null attitude, in which the three Euler angles are zero, has the OCS and BCS coincide. Null attitude is the desired attitude during normal mission mode.

The attitude knowledge requirements (the accuracy of the attitude determination) during normal mission mode are specified to be:

- 41 arc seconds in roll (3σ)
- 36 arc sec in yaw (3σ)
- 44 arc sec in pitch (3σ)

Sources of Attitude Error

When EOS-AM1 is in normal mission mode, the following quantities influence the sequential filter attitude error:

- IRU (a.k.a. gyros)
  - Rate bias errors (deg/sec)
  - Scale factor errors (dimensionless)
  - Alignment errors (deg)
  - Gyro noise such as
    a) Inverse gyro bias noise time (1/sec)
    b) Attitude error vector noise (deg/sec$^{1/2}$)
    c) Gyro bias noise (deg/sec$^{3/2}$)

- CCDSTs
  - Alignment errors (deg)
  - Measurement noise (deg)
  - Field of view errors

- FSS
  - Alignment errors (deg)
  - Measurement noise (deg)
  - Field of view errors

- Kalman filter tuning parameters. Same units as:
  - Attitude error vector noise ($deg/sec^{1/2}$)
  - Gyro bias noise ($deg/sec^{3/2}$)

THE ADEAS MODEL

What ADEAS Can Do

ADEAS is capable of modeling, on option, either the batch weighted least-squares filter or the sequential filter with Kalman gain. The latter mode is chosen for the analyses presented here. The various gyro noise parameters, CCDST and FSS measurement noises, and Kalman filter tuning parameters are user-input and are held constant for each run of ADEAS. The attitude determination uncertainty is always solved for; an initial (a priori) attitude uncertainty is specified at the beginning of the run and is usually chosen to be large to permit the filter to properly converge by avoiding numerical instabilities.

The IRU rate bias, scale factor and alignment errors, CCDST alignment and field-of-view (FOV) errors, and FSS alignment and FOV errors, can either be held at constant value (i.e., not solved for) or solved for, given a set of a priori starting values. Quantities referred to as "consider" parameters are held at constant value; the term "perfect" is sometimes used in this paper to denote a consider parameter with a value of zero (no error). Conversely, "solve-for" parameters evolve with time (as more measurements are made) and use the a priori values as initial estimates for the parameters.

This report is primarily concerned with evaluating the influence of IRU biases and scale factors and IRU, CCDST and FSS misalignments on the attitude uncertainty. The CCDST and FSS FOV errors are "perfect." (FOV errors are errors resulting from component alignments within the sensor, optical distortions, and manufacturing aberrations.)

Orbit Model Used

For purposes of internal propagation, the model orbit has the following Keplerian orbital elements, which satisfy a 10:30 a.m. mean local time descending node (Reference 1):

Epoch	980630.040000
Semimajor axis	7083.14 km
Eccentricity	0.0001
Inclination	98.2 deg
Right ascension of ascending node	255.356 deg
Argument of perigee	90.0 deg
Mean anomaly	270.0 deg

All orbit perturbative forces that can be modeled by ADEAS are enabled. These forces include the Earth oblateness J2 effect, solar and lunar point mass perturbations, and atmospheric drag with a spacecraft ballistic coefficient of 2.2.

Star Catalog Used

A prototype spectral response curve (color index) for the Ball CT-601 solid state star tracker was obtained from the Submillimeter Wave Astronomy Satellite (SWAS) project. This curve is necessary to convert star catalog visual (V) magnitudes into instrumental (I) magnitudes.

The prototype SWAS spectral response curve is

$$V - I_{SWAS} = 0.0043S^3 - 0.0015S^2 + 0.0214S - 0.1733$$

where S is the spectral index. For the Sun, a spectral class G2 star, S is equal to 4.2.

The source catalog for creating the prototype EOS run catalog is the SKYMAP Master Catalog version 3.7, a sequential file that contains approximately 248,000 stars. The SKYMAP library routine CAT then uses the prototype SWAS spectral response curve to convert V-magnitudes into I-magnitudes. It then assembles an EOS specific intermediate run catalog in a direct-access format containing stars brighter than I-magnitude 9. Intermediate catalogs are created taking into account the following criteria for each star:

- Limited to I magnitudes ranging from 2 to 6
- Excludes stars with V-magnitude uncertainties greater than 0.1

- Excludes variable stars with V-magnitude amplitudes greater than 0.1
- Excludes multiple star systems with the two brightest components having V-magnitude differences less than 5.0
- Excludes stars with proper motions greater than 0.7 arc sec per year
- Excludes stars with position uncertainties greater than 3.0 arc sec
- Uses near neighbor checks such that no star is within 0.25 degree and is within 3 I-magnitudes of the candidate star

The final sequential catalog contains 2197 stars in an ASCII-readable format. Another catalog was then produced in direct-access MMS star record format by using the SKYMAP Library routine CAT with the ASCII catalog as input. This direct access catalog is the file that is actually used by ADEAS for EOS-AM1 attitude error analysis studies. Further details on the creation of this catalog, and its comparison to the EOS star catalog created in 1990 by General Electric, can be found in Reference 2.

Alignment Angles

Reference 3 is the primary source of sensor parameters presented here. Reference 4 only slightly modifies the FSS performance requirements into a form that is identical to those for the Upper Atmosphere Research Satellite (UARS) (Reference 5).

All sensor boresights point in the +Z direction for each respective sensor coordinate system. Euler rotations in the 3-1-3 sequence transform from the BCS coordinates into each sensor coordinate system. Table 1 shows the rotation angles and Table 2 shows boresight unit vectors expressed in BCS coordinates.

Table 1. Sensor Euler Rotation Angles

Sensor	1st Rotation Z_B axis	2nd Rotation $X_{B'}$ axis	3rd Rotation $Z_{B''}$ axis
CCDST 1	-44.0	-112.5	0.0
CCDST 2	44.0	-112.5	0.0
FSS	134.299	-149.3	0.0

Table 2. Sensor Boresight Vectors (BCS)

Sensor	X	Y	Z
CCDST 1	0.6418	0.6646	-0.3827
CCDST 2	-0.6418	0.6646	-0.3827
FSS	-0.3651	-0.3563	-0.8601

Charge-Coupled Device Star Trackers

The two Ball CT-601 CCDSTs are mounted symmetrically about the BCS Y-Z plane, such that a star that appears in the FOV of CCDST1 will, in the normal mode of the attitude control system, appear in the FOV of CCDST2 after about one-third of an orbit.

Each of the CCDSTs has an 8-deg by 8-deg FOV, is sensitive to stars with I-magnitudes from +2.0 to +5.7, and can track five stars simultaneously. The major constraint to tracking to specification for these CCDSTs is that all stars within a 0.25 deg radius of a catalog star need to be at least 3 magnitudes dimmer. The CCDSTs are unreliable for the Sun within 45 deg of the CCDST boresight, or for the Moon within 17 deg of the boresight.

Current understanding of the CCDST has it raster scan from "top" to "bottom" until the first five guide stars are encountered. This approach differs from the two ADEAS CCD options: (1) choosing the five brightest stars in the FOV and (2) performing a spiral scan about the boresight until five stars are encountered. Since 4-pi steradians equals 41252.9 deg., the sky is covered by 645 CCD FOVs, for an average of 3.4 stars if the prototype EOS catalog with 2197 stars is used; so on average the chosen ADEAS option is unimportant, and option 1 is used arbitrarily. Table 3 has the required CCDST performance.

Table 3. CCDST Performance Values

	Magnitude +2.0 to +4.0	Magnitude +4.0 to +5.7
Position Accuracy 3σ, w.r.t. mounting	10 arc sec	16 arc sec
Magnitude Accuracy 3σ	0.25	0.5
Noise Equivalent Angle (NEA) 1σ	3.0 arc sec	5.0 arc sec

Here it is assumed that the position accuracy and NEA are on a per axis basis. The star catalog position uncertainty (Reference 6, per axis, 3σ) is taken to be 3 arc sec.

One ADEAS input is the standard deviation of measurement noise, which is computed for dim and bright stars, depending on whether a star is less than or greater than I-magnitude 4 in brightness. The standard deviation of measurement noise is calculated by taking the star tracker position accuracy, noise equivalent angle, and star catalog position accuracy in quadrature.

Standard deviation of measurement noise (bright stars, 3σ)

$$= [10^2 + (3 \times 3)^2 + 3^2]^{1/2} = 13.78 \text{ arc sec}$$
$$= 6.683 \times 10^{-5} \text{ rad}$$

Standard deviation of measurement noise (dim stars, 3σ)

$$= [16^2 + (3 \times 5)^2 + 3^2]^{1/2} = 22.14 \text{ arc sec}$$
$$= 1.073 \times 10^{-4} \text{ rad}$$

The 3σ alignment uncertainty, per axis, is 0.05 deg.

Fine Sun Sensor

The Adcole FSS (Model 42050 sensor and Model 42070 electronics) is composed of two orthogonally mounted single-axis Sun sensors. Each single-axis Sun sensor consists of two reticles: a fine reticle and a coarse reticle. The coarse reticle pattern is gray-coded and encodes the coarse angle over the entire FOV. The fine reticle patterns and the resultant photocell currents are used to generate fine-angle data. The overall FOV is 64 deg square. The output resolution is 14 arc sec per least significant bit. The overall accuracy for a 32-deg half-cone is 60 arc sec, and the accuracy between the half-cone of 32 deg and the FOV of ±32 deg is 120 arc sec.

As the above description also applies to the UARS FSS, we may extract the standard deviation of measurement noise from Reference 5, which gives the value 75 arc sec or 0.02083 deg for 3σ uncertainty. The 3σ alignment uncertainty, per axis, is 0.05 deg.

Inertial Rate Unit

The Kearfott IRU is composed of three independent channels, each channel having one two-axis gyro and associated electronics. Under command, the IRU is sensitive to either of two rate ranges:

- Low Rate: ±0.11 deg/sec maximum rate, with a scale factor of 0.05 arc sec/pulse for incremental output
- High Rate: ±2.0 deg/sec maximum rate, with a scale factor of 0.8 arc sec/pulse for incremental output

The analog rate output range is ±2.0 deg/sec.

ADEAS does not model low or high IRU rates, or the analog output, since ADEAS uses engineering units internally, rather than counts or pulses.

The following 3σ uncertainties are allocated per axis:

- Noise (white) 0.0001 deg/sec$^{1/2}$
- Noise (drift) 0.00138 deg/hr$^{3/2}$ $= 6.3889 \times 10^{-9}$ deg/ sec$^{3/2}$
- Alignment 0.1 deg
- Bias 2.0 deg/hr $= 5.5555 \times 10^{-4}$ deg/sec

From Reference 5, the 3σ uncertainty in the standard deviation of the scale factor, per axis, is 1.4×10^{-5} (UARS value). The inverse gyro bias noise time constant is assumed to be 0.0, an appropriate value for white noise.

Kalman Filter Parameters

The Kalman filter parameters were chosen to have the same values as the attitude error vector noise (white noise) and gyro bias noise (drift noise) tabulated for the IRU above. No attempt was made in this analysis to tune these parameters for optimum convergence.

ESTIMATED SENSOR UNCERTAINTIES

Methodology

Numerous runs of duration 6000 sec (slightly longer than one orbital period) were performed, with CCDST and FSS sensor data simulated at 10-sec intervals. Various combinations of solve-for and consider sensor parameters were used, with a priori and consider values taken from the prelaunch errors of Table 4 below. CCDST1 was assumed to be "perfect" and was therefore the reference coordinate system for the calibrations. The initial attitude uncertainty was set to 999.0 deg to ensure that the starting attitude knowledge was unknown.

Throughout this section and the next, two systems of units are used, the ADEAS inputs/outputs (in degrees and degrees per second) and units more suitable for interpretation and comparison with mission requirements (arc seconds and arc seconds per hour).

Table 4. Prelaunch Uncertainties (3σ)

Sensor	Quantity	Value	Equivalent Value
Inertial Reference Unit (IRU)	Rate Biases	5.555E-4 deg/sec	7200 arc sec/hr
	Scale Factor	1.4E-5	n/a
	Misalignment	0.1 deg	360 arc sec
CCD Star Trackers (CCDST)	Misalignment	0.05 deg	180 arc sec
Fine Sun Sensor (FSS)	Misalignment	0.05 deg	180 arc sec

Calibration results for CCDST2 and the FSS did not improve (results were much larger than the prelaunch values, by up to a factor of 3), and the attitude uncertainties would not shrink below about 0.3 deg, unless IRU rate biases were solved-for simultaneously. This empirical observation makes sense. During periods when no sensor data are available, the attitude is propagated using the IRU bias, the fixed bias uncertainty, and noise. When the CCDST2 and FSS do have data, their observation vectors conflict with that expected from the dynamically modeled attitude, and the CCDST2/FSS alignment errors grow to compensate. The necessity to continually solve for IRU biases was independent of the size of the bias uncertainties when specified as consider parameters, as the computed CCDST2/FSS alignment errors remained large, as did the attitude uncertainties.

Attempting to solve for all parameters at once took an inordinate amount of time, and the IRU scale factor uncertainties did not change at all. Thus, calibration runs solved for IRU rate bias errors and IRU, CCDST2, and FSS alignment errors.

Calibration runs were performed for 14 different attitude maneuver scenarios. Table 5 below lists the details of three schemes. Maneuver 1 is the nadir-pointing 1 rpo case, which is the nominal attitude profile. Maneuver 2 is a ±5 deg roll offset from nominal, not unlike what UARS used for its on-orbit calibration. Maneuver 3 is a ±20 deg roll offset version of Maneuver 2. The table indicates the attitude angles and rates at the beginning and end of the run, along with times at which new attitude rates are commanded and the attitude offsets at those times.

Table 5. Attitude Maneuver Profiles

Maneuver # and Description	Time t into run (sec)	Roll (deg)	Roll rate (deg/sec)
Maneuver 1 (1 rpo)	0	0.0	0.0
	6000	0.0	0.0
Maneuver 2 (±5 deg roll)	0	0.0	0.0
	1800	0.0	8.33E-3
	2400	5.0	-8.33E-3
	3600	-5.0	8.33E-3
	4200	0.0	0.0
	6000	0.0	0.0
Maneuver 3 (±20 deg roll)	0	0.0	0.0
	1800	0.0	3.33E-2
	2400	20.0	-3.33E-2
	3600	-20.0	3.33E-2
	4200	0.0	0.0
	6000	0.0	0.0

CCDST and FSS Alignment Uncertainties

The computed results for CCDST2 uncertainties were independent of the maneuver scheme used, as one would expect. FSS uncertainties did vary between maneuver schemes, but this variation can be attributed to differing periods of Sun visibility (23 minutes for Maneuvers 1 and 2 and 20 minutes for Maneuver 3), so differing amounts of data were available for calibration. For example, the FSS uncertainties from Maneuver 3 were 20 percent larger than for Case 1 of Table 6.

The results presented in Table 6 were all derived from Maneuver 1, a nominal pointing scenario. Case 2 is identical to Case 1, except that it is a 12,000-sec run. Case 3 used the results of Case 1 as a priori uncertainties, which illustrates that repeated sensor alignment calibrations will result in smaller alignment uncertainties. Thus, longer spans of data, and accurate estimates of alignments after a calibration run, will result in subsequent calibration runs having smaller uncertainties. In principle, with sufficiently long runs, these misalignment errors can be made arbitrarily small. However, the overall sensor uncertainty will not necessarily behave similarly, since the sensor measurement noises become the dominant effect. The prelaunch alignment uncertainties are shown in Table 6 for comparison.

The X, Y, and Z components of CCDST2 and FSS alignment uncertainties need to be interpreted carefully, as they represent uncertainties in rotation angles about the nominally aligned sensor coordinate system for each sensor, with Z being the boresight vector.

Table 6. Alignment Uncertainties (arc sec, 3σ)

Case and Description	CCDST2 (arc sec, 3σ)			FSS (arc sec, 3σ)		
	X	Y	Z	X	Y	Z
Prelaunch	180	180	180	180	180	180
Case 1: 1 orbit	10.4	4.0	11.5	6.1	12.6	14.8
Case 2: 2 orbits	7.2	2.8	7.9	4.3	9.0	10.4
Case 3: use case 1 as a priori	5.4	2.1	7.9	4.3	7.2	10.4

IRU Alignment and Rate Bias Uncertainties

Table 7 is a comparison of the prelaunch IRU rate bias uncertainties and the IRU misalignment uncertainties with the uncertainties computed for the three attitude maneuver calibration schemes of Table 5.

Table 7. IRU Rate Bias and Alignment Uncertainties

Maneuver Scheme and Description	IRU Rate Bias Uncertainty (arc sec/hr, 3σ)			IRU Alignment Uncertainty (arc sec, 3σ)		
	X	Y	Z	X	Y	Z
Prelaunch	7200	7200	7200	360	360	360
No. 1 (1 rpo)	1296	21	1296	353	360	353
No. 2 ±5 deg roll	233	21	518	137	54	61
No. 3 ±20 deg roll	58	21	143	36	15	15

From Table 7 we can draw the following conclusions:

1. The IRU misalignment errors cannot be improved unless maneuvers that deviate from nominal attitude (scheme No. 1)are performed.

2. Larger calibration maneuvers result in smaller IRU alignment uncertainties.

3. Larger calibration maneuvers result in smaller IRU X- and Z-axis rate bias uncertainties.

. The IRU Y-axis rate bias uncertainty is unaffected by the magnitude of the calibration maneuver.

The latter conclusion is explained by maneuvers decreasing the magnitude of the Y-axis angular rate, whereas an increase in the rate would be required to reduce the rate bias uncertainty.

ESTIMATED ATTITUDE UNCERTAINTIES

Results Using Prelaunch Alignment Uncertainties

Attitude was solved-for using various combinations of prelaunch consider values and "perfect" values for the attitude sensors and IRU parameters. This was done to gain some appreciation of which consider parameters had the most impact upon the computed attitude uncertainty. Table 8 indicates that prelaunch consider values for the IRU biases are the greatest single contributor, followed by the CCDSTs and FSS, the IRU misalignments, and, finally, the IRU scale factors, which had an 8 arc sec level of uncertainty.

Table 8. Attitude Errors With Prelaunch Uncertainties (IRU Bias not Solved For)

Solve-for Parameters	Prelaunch Consider Values	"Perfect" (Consider=0)	Attitude Uncertainty (arc sec, 3σ)
Attitude only	IRU RBs IRU SFs IRU M/As CCD1 M/As CCD2 M/As FSS M/As		673
	IRU RBs IRU SFs IRU M/As CCD2 M/As FSS M/As	CCD1 M/As	644
	IRU RBs IRU SFs IRU M/As	CCD1 M/As CCD2 M/As FSS M/As	605
	CCD1 M/As CCD2 M/As FSS M/As	IRU RBs IRU SFs IRU M/As	292
	IRU RBs	IRU M/As IRU SFs CCD1 M/As CCD2 M/As FSS M/As	601
	IRU M/As	IRU RBs IRU SFs CCD1 M/As CCD2 M/As FSS M/As	86
	IRU SFs	IRU RBs IRU M/As CCD1 M/As CCD2 M/As FSS M/As	8
LEGEND:	RBs: Rate Biases M/As: Misalignments	SFs: Scale Factors CCD: CCDST	

Attitude and IRU biases were also solved-for, mimicking the onboard computer and its sequential filter and using prelaunch consider values for all other sensors, with differing combinations of the CCDSTs and FSS functioning. These combinations are shown in Table 9. The first case is for all three sensors "on" as a baseline, even though this is not a flight mode. The other three combinations shown are flight modes, namely CCDST1 and 2, CCDST1 and FSS, and CCDST1 by itself. As one would expect, the last case shows the largest uncertainty.

Table 9. Attitude Errors With Prelaunch Uncertainties (IRU Bias Solved For)

Solve-for Parameters	Prelaunch Consider Values Used	Sensor Combination Used	Attitude Uncertainty (arc sec, 3σ)
Attitude and IRU RBs	IRU SFs IRU M/As CCD1 M/As CCD2 M/As FSS M/As	CCD1 CCD2 FSS	292
	IRU SFs IRU M/As CCD1 M/As CCD2 M/As	CCD1 CCD2	292
	IRU SFs IRU M/As CCD1 M/As FSS M/As	CCD1 FSS	288
	IRU SFs IRU M/As CCD1 M/As	CCD1	317
LEGEND:	RBs: Rate Biases M/As: Misalignments	SFs: Scale Factors CCD: CCDST	

The root sum square (RSS) uncertainties shown in Tables 8 and 9 were for the attitude uncertainties at the end of each respective ADEAS run.

Results Using Solved-For Alignment Uncertainties

All of the results in this section were based on a sequential filter, solving for the attitude and IRU rate bias uncertainties, as does the actual EOS-AM1 onboard computer. The computed attitude uncertainties are with respect to the BCS. The attitude uncertainties based on the prelaunch uncertainties utilize the uncertainties for all sensors (CCDST1, CCDST2, FSS, and IRU misalignment errors), whereas the attitude uncertainties computed using the calibration profile based on Maneuver 1, also known as "on-orbit" nominal, assume that the CCDST1 alignment is perfectly known. The prelaunch consider values for the IRU scale factor errors were used in all cases.

The calibration profiles based on Maneuvers 2 and 3 were also applied against Maneuver 1, with no change in results, and are therefore not shown in Tables 10 through 12.

Use is also made of ADEAS' capability to display the error budget for each computed uncertainty. This shows the contribution of each consider parameter, measurement noise, and dynamic noise to the overall error.

Two CCDSTs and One FSS

This is not a flight mode for the onboard computer, but these solutions are provided as a baseline . The attitude uncertainties based on the prelaunch sensor alignment uncertainties are dominated by the CCDST1 and CCDST2 uncertainties (large), whereas the attitude uncertainties based on the 1-rpo calibration are dominated by CCDST2 misalignment uncertainties (small), closely followed by measurement noise. Use of the 1-rpo calibration profile easily satisfies mission requirements for EOS-AM1 in the nominal attitude.

Table 10. Attitude Error for CCDST1 & 2, FSS

Calibration Profile	Maneuver Profile	Attitude Uncertainty (arc sec, 3σ)		
		X	Y	Z
Mission Requirement	N/A	41.0	44.0	36.0
Prelaunch Values	Maneuver 1 (1 - RPO)	166.0	184.0	155.0
On-Orbit (1-RPO)	Maneuver 1 (1-RPO)	8.3	9.0	5.4

Two CCDSTs

The nominal sensor complement for onboard attitude determination is two CCDSTs. These numbers do not differ at all from those of Table 10, and the error budgets are identical. Thus the conclusions for two CCDSTs are identical to those presented for two CCDSTs and the FSS, except that the FSS would not improve the attitude solution. This is as one would expect intuitively.

One CCDST and One FSS

This is the first contingency mode for the onboard computer, in case one CCDST fails. The attitude uncertainties based on the prelaunch sensor alignment uncertainties are dominated by the CCDST1 uncertainties (large), closely followed by the FSS uncertainties. The attitude uncertainties based on the 1-rpo calibration are dominated by measurement noise. Use of the 1-rpo calibration profile easily satisfies mission requirements for EOS-AM1 in the nominal attitude, and the error is only slightly worse than the cases with two CCDSTs.

Table 11. Attitude Error for CCDST1 & FSS

Calibration Profile	Maneuver Profile	Attitude Uncertainty (arc sec, 3σ)		
		X	Y	Z
Mission Requirement	N/A	41.0	44.0	36.0
Prelaunch Values	Maneuver 1 (1 - RPO)	176.0	155.0	166.0
On-Orbit (1-RPO)	Maneuver 1 (1-RPO)	11.5	12.2	9.7

One CCDST

This is the second and last contingency mode for the onboard computer, in case one CCDST and the FSS both fail. The error budgets for the various cases parallel those listed in the previous section, ignoring all references to the FSS. Again, use of the 1-rpo calibration profile easily satisfies mission requirements for EOS AM-1 in the nominal attitude. Notice that the lack of FSS measurements did not change the level of uncertainty from the CCDST/FSS scenario.

Table 12. Attitude Error for CCDST1 Only

Calibration Profile	Maneuver Profile	Attitude Uncertainty (arc sec, 3σ)		
		X	Y	Z
Mission Requirement	N/A	41.0	44.0	36.0
Prelaunch Values	Maneuver 1 (1 - RPO)	180.0	180.0	180.0
On-Orbit (1-RPO)	Maneuver 1 (1-RPO)	11.5	12.2	9.7

CONCLUSIONS

Relative to one star tracker, the alignment uncertainties of the other star tracker and the fine Sun sensor become smaller asymptotically with the use of longer spans of sensor data. These results are independent of the attitude maneuver scenario, except for the maneuver's influence upon the length of time that the Sun is visible to the FSS.

Attitude maneuvers that deviate from nominal pointing are necessary for improving the IRU alignment uncertainties. The larger the attitude maneuvers, the smaller the IRU alignment uncertainties, which in turn reduces the IRU rate bias uncertainties. The smaller the IRU rate bias uncertainties, the more accurate will be the propagated attitude solution in the onboard Kalman sequential filter during those times when sensor observations are unavailable. The statements of this and the preceding paragraph, although derived from the particulars of the EOS-AM1 mission, are true independent of the details of a particular satellite.

Since performing a ±20 deg roll maneuver reduces the IRU alignment uncertainties by a factor of 10 or more, and reduces the IRU rate bias uncertainties by a factor of 50 or more (in both cases relative to the prelaunch uncertainties), it is recommended that attitude calibration maneuvers have at least a 20 deg excursion from the nominal 1-rpo attitude profile and, if possible, that they include both roll and yaw maneuvers.

If no calibrations are performed, and the EOS-AM1 sequential filter is used to solve for the attitude and IRU rate biases, then the RSS absolute attitude uncertainty is of the order of 300 arc sec, which is far in excess of the mission requirements. Therefore, some attempt at calibration must be made. A calibration profile solely based upon the nominal 1-rpo motion will suffice for all sensor combinations.

As suggested by data presented in this report, calibration maneuvers should have a minimum of 20 deg in roll and yaw, which would provide robustness to the accuracy of attitude solutions for large deviations from nominal 1-rpo pointing. Previous analyses for other missions have indicated that 30-deg maneuvers about each axis are needed to achieve the best results. As the EOS orbits will have plenty of star observations for attitude determination, the choice of one CCDST as backup to the two-CCDST configuration is adequate. The FSS would not improve the attitude solution unless star observations were unavailable due to Sun/moon interference in the CCDST, during times when the Sun is visible to the FSS.

The results of this analysis show that the onboard attitude determination function will be more than able to meet the uncertainty requirements.

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