Ground Control of Drag Satellites Utilizing Onboard Accelerometer Data

A. J. Fuchs* and C. E. Velez†
NASA Goddard Space Glight Center, Greenbelt, Md.

High drag satellites frequently require precise vertification of orbital maneuvers and the accurate prediction of perigee height. An inflight ground support system designed to monitor and compute orbital state and maneuvers is described. The use of onboard three-axis accelerometer data in a flight support software system to perform online maneuver analysis and atmospheric model updating is discussed. In addition, automated analytic techniques to predict perigee height rapidly and accurately following a maneuver are described, as well as semianalytic averaging techniques designed to predict a decaying orbital state for mission control.

Introduction

SATELLITES that penetrate the Earth's atmosphere in the neighborhood of 100 to 200 km frequently require precise verification of orbital maneuvers and the accurate prediction of perigee heights. One method of performing these functions is by careful analysis of many orbit determination solutions obtained from the reduction of precision tracking data. An alternative approach that has recently been receiving increased attention is through the utilization of onboard accelerometer data.

This paper will present some results of in-flight ground control of the Atmosphere Explorer (AE-C) satellite utilizing accelerometer data. In addition, some rapid orbit prediction techniques used to predict perigee heights and to monitor the evolution of the orbit for mission planning purposes are described. The first section presents background information on the Atmosphere Explorer project and a description of the onboard accelerometer package. The following section presents the flight support software system, including the accelerometer processor and filtering routines, designed for AE-C.

Following this description, specific results obtained during the initial months of AE-C support are presented. Maneuver verification obtained from the accelerometer data is compared to verification obtained from the reduction of precision tracking data. It is seen that, when real-time decisions are required, the accelerometer data provide the more expedient means of obtaining accurate results. A calibration of various atmospheric models used in the Goddard Space Flight Center orbit determination and orbit prediction programs is made by comparing predicted drag accelerations with measured drag accelerations over the same time span. Algorithms are developed to utilize the accelerometer data 1) to replace components of the forcing function in the orbit generator and 2) to estimate critical parameters of the mathematical drag model by implementing the measured accelerations as an observational data type. Finally, an analysis of the attitude content of the data, rapid orbit prediction schemes used on AE-C, and some potential applications of accelerometer data are discussed.

Mission and Accelerometer Background

The Atmosphere Explorer-C (AE-C) satellite was launched on Dec. 15, 1973 from the Western Test Range by a two-stage thrust-augmented Delta launch vehicle. The objective of this project is to investigate the photochemical and energy transfer processes that take place in the largely unexplored low-altitude region between 120 and 400 km in the earth's atmosphere.[1] The spacecraft was launched into an inclined, eccentric orbit whose initial approximate elements are shown in Table 1.

The spacecraft contains an orbit adjust propulsion system (OAPS), which is used to alter the apogee and perigee heights of the orbit. During phase one of the mission (which will last approximately eight months), perigee will be alternately lowered and raised between 150- and 120-km altitudes in steps while maintaining apogee at approximately 4300 km. During phase two of the mission, the spacecraft will be placed into a series of circular orbits ranging from 250 to 800 km.

Following each perigee adjustment, a computation requirement exists to determine the new perigee height accurately within 15 min following the maneuver, such that remedial action may be taken if the perigee height is hazardously low. The perigee adjustments, of course, take place while the spacecraft is at apogee so that a trajectory determination and propagation are required to determine minimum altitude. In addition, a computational requirement exists to provide accurate trajectory and mission profiles to the aeronomy team for periods up to two months so that experiments and correlative measurements may be made at specific times.

One of the major experimental packages onboard the spacecraft is an atmosphere density accelerometer designed to determine the density of the neutral atmosphere in the region between 120 to 400 km. The experiment consists of three uniaxial miniature electrostatic accelerometer (MESA) units or proof masses suspended by electrostatic forces and rebalanced (constrained) along its sensitive axis by electrostatic forces.[2] The MESA units are contained in a cylinder 13 in. long and 10 in. in diameter, mounted at the center of the satellite. The sensitive axes of the units are mounted to form an orthogonal triad (*XY, YX, Z*). The *Z* accelerometer lies along the nominal spin axis of the spacecraft while the other two lie in the perpendicular *XY* plane ±45 deg from the *X*-

Table 1 AE-C initial orbital elements

Apogee altitude (km)	4306.00
Perigee altitude (km)	156.00
Semimajor axis (km)	8610.00
Eccentricity	0.24
Inclination (deg)	68.12
Right ascension of ascending node (deg)	253.09
Argument of perigee (deg)	165.64
Mean anomaly (deg)	203.28
Epoch Dec. 17, 1973	8 hr 00 min 30 sec UT

Presented as Paper 74-809 at the AIAA Mechanics and Control of Flight Conference, Auabeim, Calif., August 5-9, 1974; submitted October 31, 1974; revision received September 19, 1975.

Index categories: Spacecraft Navigation, Guidance, and Flight-Path Control Systems; Earth-Orbital Trajectories.

*Head, Orbital Mechanics Section.

†Head, Systems Development and Analysis Branch. Member AIAA.

axis of the spacecraft. Secondary objectives of the experimental package are to monitor thrust during orbit adjust propulsion system operations, to determine satellite minimum altitude, and to provide some attitude sensing information.

The following sections will describe the software components designed to utilize the onboard accelerometer data to satisfy the computational requirements just described and to satisfy the secondary objectives of the experiment. Also, numerical results on applications of the MESA will be presented.

Flight Support System

The flight support system for AE-C was designed in conformance with the flight dynamics system (FDS) configuration at Goddard Space Flight Center (GSFC). The system consisted of orbit determination, attitude determination, orbit adjust maneuver, and accelerometer data processing programs. Interfaces existed between each of these subsystems to support the accelerometer data modules for real-time maneuver verification and drag model calibration.

Initially the orbit adjust maneuver program (OAMP) is used to plan a maneuver, given an orbit change requirement and an estimate of the orbit state. For a perigee raising or lowering maneuver, this program computes tables of expected changes in perigee height and velocity during the maneuver for later comparison with actual MESA telemetry. During the actual maneuver, the thrust accelerations are measured via the MESA, which records average acceleration values over a quarter-second sample time and are transferred sequentially through an analog to digital converter to a buffer region from where they are transmitted to a ground receiving station in the form of pulses. The accelerometers are reset every quarter second. The pulses are telemetered along with all other telemetry (TM) data from the receiving station to GSFC. At Goddard the data is routed to the control center Sigma 9 computer and to the IBM 360 computer for use by the attitude determination program and the MESA processor. The attitude system separates the MESA data from the other TM data and passes it on in nominal records of 56 sec. The data are nominally available to the MESA processor one minute following engine ignition.

The MESA processor displays comparisons between the OAMP predicted perigee height and velocity increments and those obtained from the MESA data. The orbital state at the initiation of burn is relayed to the MESA processor via the orbit determination program.

The acceleration a_p sensed by a MESA is related to the actual acceleration $a(t)$ by

$$a(t) = s(a_p + b) + v \tag{1}$$

where a_p is the pulse measurement, s is a temperature-dependent scaling parameter, b is a sensor-dependent bias, and v is measurement noise.

Upon receipt of a record of MESA data, zero-set biases are removed and temperature-corrected scaling factors are applied to convert the pulses to engineering units. The biases and scaling factors are determined in orbit, by procedures given in Ref. 2. In addition, data dropouts are filled by using smoothing polynomials. The acceleration polynomials are twice integrated, as shown by Eqs. (2) over discrete times during the maneuver and the resultant position and velocity increments added to the orbital state sequentially in an Encke scheme for propagation and evaluation of changes to the orbital state. The propagation equations are

$$a_t = \sum_{i=0}^{N} b_i t^i \tag{2a}$$

$$\Delta V_t = \int_0^t \sum_{i=0}^{N} b_i t^i \tag{2b}$$

$$\Delta r_t = \int_0^t \int \sum_{i=0}^{N} b_i t^i \tag{2c}$$

where a_t is the acceleration over the interval t, N is the order of the polynomial, b_i [2-5] are the polynomial coefficients determined from least squares fits to the data, and ΔV_t, Δr_t are the position and velocity increments, respectively, which are added stepwise to a semianalytic integration of two-body and oblateness accelerations. A complete description of the MESA processor is given in Ref. 3.

When the spacecraft passes through the atmosphere (through perigee of the orbit), the measured accelerations in the form of pulses are recorded for subsequent playback when in view of a ground station. During a perigee pass the spacecraft is maintained either in a spinning or despun mode, and when spinning, the MESA rotates with the spacecraft.

The acceleration $\bar{a}_t$ at time t on the proof mass located at the top of a vector $\bar{r}$ from the center of gravity of the spacecraft will be given after scaling and bias removal by

$$\bar{a}_t = \underline{\Omega}(t)\bar{a}_F(t) + \dot{\bar{w}} \times \bar{r} + \bar{w} \times \bar{w} \times \bar{r}) + \bar{\epsilon} \tag{3}$$

where $\bar{a}_F(t)$ represents the sum of all external forces in inertial coordinates, $\bar{w}$ is the spacecraft angular velocity vector, $\bar{\epsilon}$ is a noise component due to any other motion of the spacecraft, and $\Omega(t)$ is the Eulerian inertial to body-fixed rotation matrix.

Thus, when the spacecraft is spinning, the external acceleration is modulated by a time-varying rotation matrix in addition to the presence of coriolis and centripetal accelerations and an unmodeled component due to other spacecraft motion. It is desirable to extract from this acceleration a "pure" measurement of the external force (drag). This is most easily accomplished by use of a digital filter.[4,5] This is made possible by the fact that the signal representing the effects of the external accelerations is contained in a frequency band which differs significantly from the frequency bands due to spacecraft motion and noise sources. A filter designed to pass the drag frequency and reject all other undesired frequencies will yield the proper accelerations. Reference 4 contains a detailed description of the filter designed for AE-C.

The filter design for AE-C was based upon many factors, of which some of the more important are: frequency response, time delays, and computational efficiency. The filter chosen was a cascaded version of the Butterworth class, characterized by placement of the poles and zeroes in the complex z-plane, resulting in a flat passband and sharp rolloff characteristics.

Figures 1-3 show the raw accelerometer data received during a perigee pass on AE-C on March 11, 1974 with a perigee height of approximately 145 km. Figure 4 depicts the response of a band-pass Butterworth filter centered about the spin frequency of the spacecraft, to the XY data. The YX response is similar to XY and the Z component is two orders of magnitude smaller than the other two components. The spin frequency was deduced by performing a power spectral density analysis on the measurement data. Figure 5 is a root sum square of the three filtered components which, due to the orthogonality of the sensors, yields the measured acceleration estimate.

Applications of MESA

By the end of April, 1974, the accelerometer data had been used to support some 20 perigee altitude adjustments on AE-C. An example of accelerometer data received during a perigee lowering maneuver is shown in Fig. 6. Readily observable from these data are zero-set biases, noise levels, thrust buildup, etc. The following discussion is a brief summary of the perigee height predictions obtained during the maneuver sequency from March 6, 1974 through March 11, 1974. Table 2 shows the sequence of desired perigee heights, together with the perigee heights determined from the MESA data and from the post-maneuver tracking data. The perigee

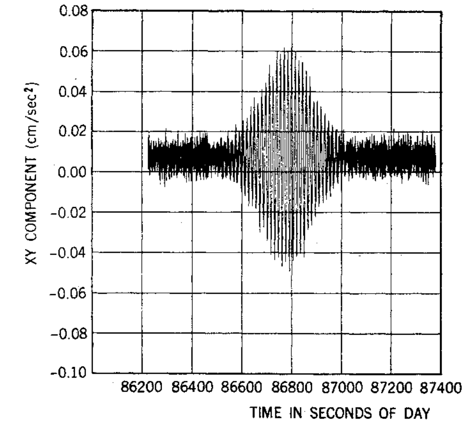

Fig. 1 AE-C XY raw accelerometer data, March 11, 1974 (145 km).

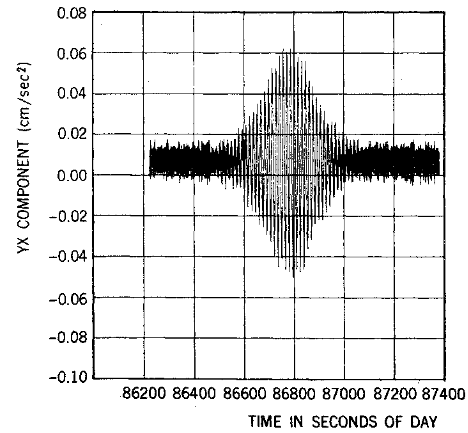

Fig. 2 AE-C YX raw accelerometer data, March 11, 1974 (145 km).

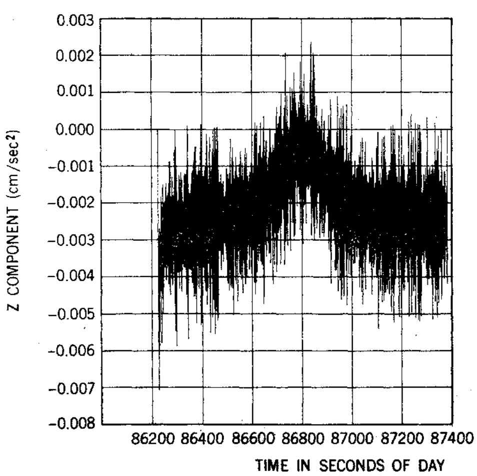

Fig. 3 AE-C Z raw accelerometer data, March 11, 1974 (145 km).

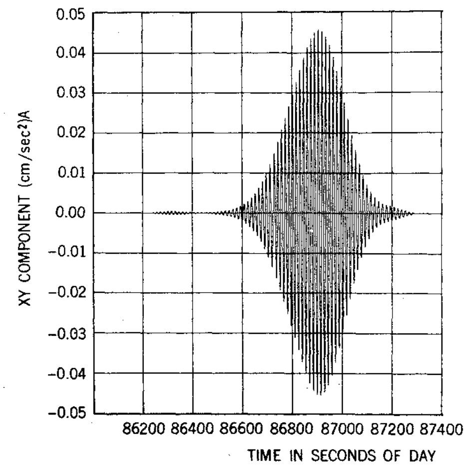

Fig. 4 AE-C filtered XY data, March 11, 1974 (145-km perigee).

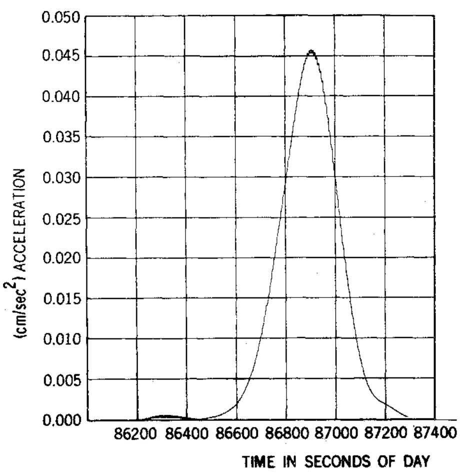

Fig. 5 AE-C total measured acceleration, March 11, 1974 (145-km perigee).

heights determined from the MESA data were computed immediately following the maneuver (within 3 min) by integrating the accelerations received via telemetry data, adding the velocity increments to the orbit state at that time, and predicting the resultant orbit to perigee. The perigee heights computed from tracking data utilized data following the maneuver. These perigee height predictions were generally available 15 min following engine cutoff. The percent differences between the targeted perigee and those computed from MESA and tracking data were consistently less than one percent. The percent differences between the MESA and track solutions exhibit no significant trends and confirms the independence of these solutions for maneuver confirmation.

Table 2 Perigee height predictions

Target perigee height(km)	Measured from MESA(km)	Measured from tracking data(km)	Desired MESA (%)	Desired track (%)	MESA track (%)
150	150.2	149.6	−0.1	+0.3	+0.4
145	–	145.0	–	0.0	–
150	149.5	150.9	+0.3	−0.6	−0.9
145	145.2	144.3	−0.1	−0.1	+0.6
140	140.2	139.9	−0.1	+0.1	+0.2
145	144.7	144.9	+0.2	+0.1	−0.1
140	–	140.1	–	−0.1	–
148	147.9	147.9	+0.1	+0.1	0.0

Table 3 Velocity magnitude changes from OAMP and MESA

Target perigee (km)	Predicted velocity Magnitude change (cm/sec)	Velocity magnitude Change from MESA (cm/sec)	% difference Predicted MESA
150	296.1	291.9	+ 1.4
145	119.2	–	–
150	128.9	114.9	+12.2
145	152.1	148.3	+ 2.6
140	110.5	110.6	−0.1
145	129.1	127.2	+ 1.5
140	–	–	–
148	203.4	201.0	1.2

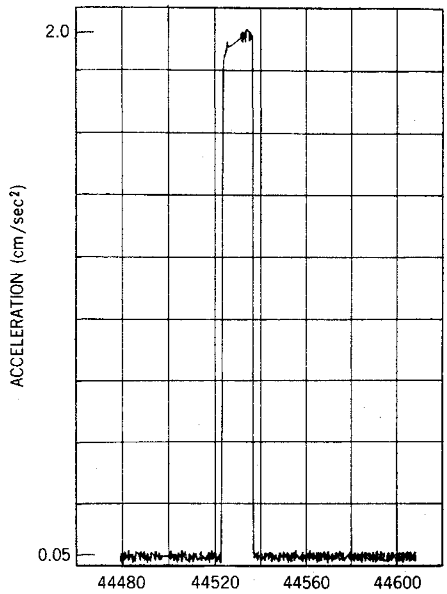

Fig. 6 AE-C XY sensor despun burn.

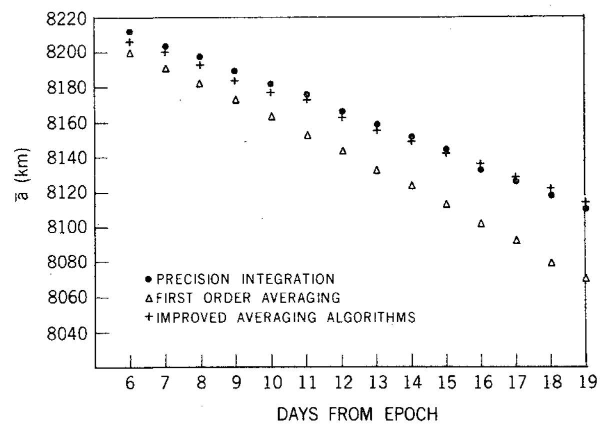

Fig. 7 AE-C semimajor axis decay history.

Table 3 shows the velocity magnitude changes as computed by the MESA and as predicted by the OAMP program. Since the onboard engines actually cut off on accumulated velocity changes, this table provides a performance estimate of the MESA program calculations. Except for the third maneuver, all estimates are within 3% of the actual values.

Table 4 tabulates the burn durations as predicted by OAMP and as calculated from the MESA data. Barring data dropouts at the start and termination of the burns (which were negligible for all maneuvers of this sequence) the MESA calculations are accurate and provide estimates of engine performance. The constant bias in the percent differences between the predicted and MESA calculated burn durations indicates either a bias in the ground calibration constants or engine performance.

The rapid and accurate prediction of orbit state following maneuvers is a necessary step in maneuver verification. As already mentioned, state estimates following maneuvers are available from the integration of accelerometer data or from the reduction of tracking data. In either case an osculating post-maneuver orbit is available. An automated analytic technique to propagate the state to future events (such as perigee) is shown. The technique uses analytical expressions for the short periodic variations in the elements (due to Brouwer-Ref. 8) as follows: e(perigee) = e(post-maneuver) − Δe(post-maneuver) + Δe(perigee), where e represents the osculating elements and Δe the short periodic variations to these elements as given in Ref. 8, evaluated first at the post-maneuver time and then at perigee. The elements at perigee are then used to calculate the minimum height about the oblate Earth. The accuracy of the above technique is within one percent of the precision integration to perigee, yet requires negligible computer time.

A semianalytic technique to propagate the evolution of the orbital elements over longer periods of time makes a simple modification [6] to the classical averaging technique as follows:

$$\dot{\bar{e}} = \frac{1}{\tau(\bar{e})} \int_{t-\tau/2}^{t+\tau/2} f(\bar{e} + \Delta e + \dot{\bar{e}}'s, s)\,\mathrm{d}s$$

where $\bar{e}$ represents the mean orbital state, τ the orbital period derived from the "mean" mean motion, Δe the first-order short periodic corrections, and $\dot{\bar{e}}$ the secular rates obtained at the previous integration step. Results for the propagation of the AE-C semimajor axis comparing this "improved" averaging with the classical averaging and precision integration are shown in Fig. 7.

The accelerometer data are also utilized to calibrate atmospheric density models based upon data received during perigee passes. Data for two such passes at 155 km and 145 km are shown in Fig. 8 and 9, respectively. The atmospheric models used in the comparison are the Jacchia-Roberts 1971,

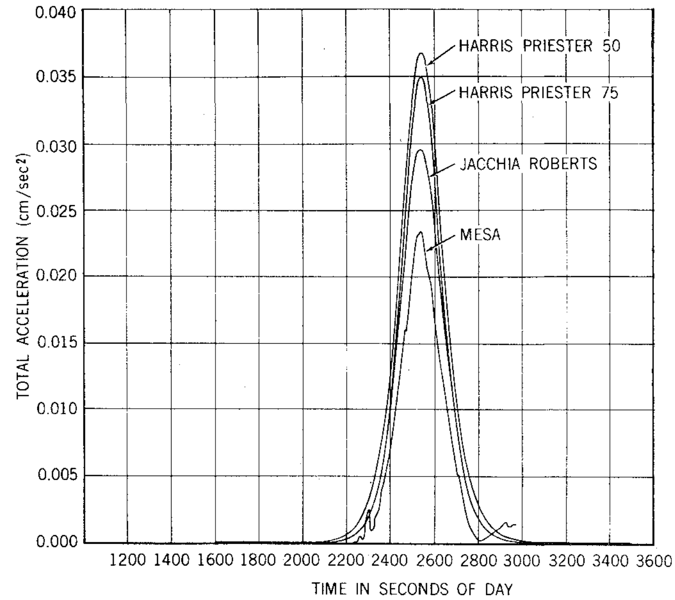

Fig. 8 Model and measured drag accelerations (155-km perigee).

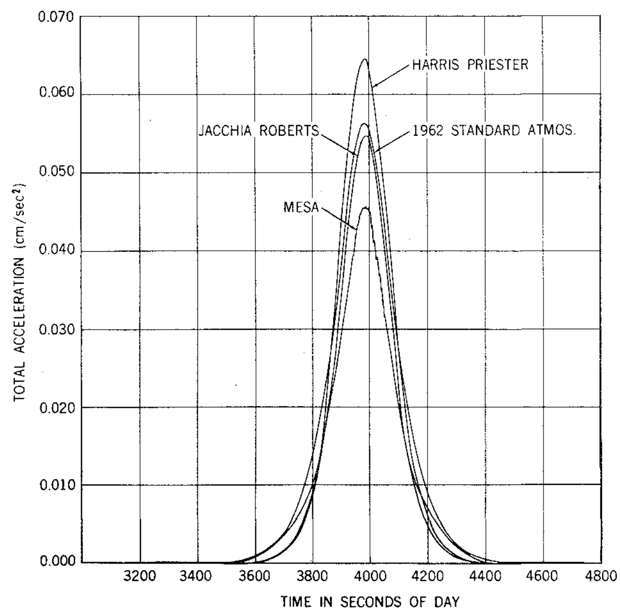

Fig. 9 Model and measured drag accelerations (145-km perigee).

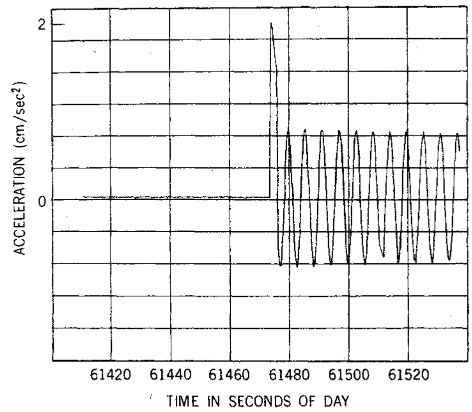

Fig. 10 AE-C XY sensor yaw maneuver.

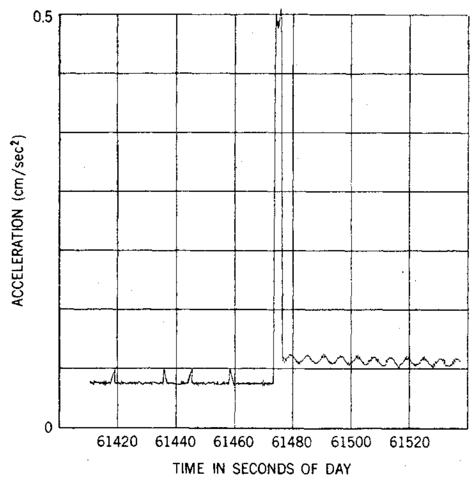

Fig. 11 AE-C Z sensor yaw maneuver.

model, the Harris-Priester model, and the 1962 standard atmospheric model. The percent errors at the peak acceleration vary from approximately 20% for the Jacchia-Roberts, which agrees with results in Ref. 7 for data at higher altitudes, to over 40% for the Harris-Priester model.

The information content of the accelerometer data relating to spacecraft attitude is readily observable from Figs. 10 and 11. The time span covered by the data is before, during, and after a yaw maneuver performed to alter the spin period of the spacecraft. The change in spin frequency can be observed from both the XY and Z axis, remembering that the Z axis will measure the centripetal acceleration. In addition, a very slow nutation damping may be seen in Fig. 11.

Potential Applications and Goals

A great deal of work is planned in the area of evaluating the ability to use the accelerometer data to improve the orbit predictions of drag perturbed satellites.

An experimental version of the orbit determination system has been developed which will allow for utilizing the MESA data to: 1) Replace the mathematical models for atmospheric drag and thrust accelerations. 2) Estimate critical parameters in the existing atmospheric models by using the MESA data as an observation type along with tracking data.

This program (already developed) will be the primary tool used to evaluate the ability of using the accelerometer data to improve orbital predictions for satellites where atmospheric drag represents the largest source of uncertainty. The two functions will be evaluated independently.

The model replacement function has potential application for onboard computer usage (on such satellites as the Solar Maximum Mission and Drag Free Satellites) actually to

Table 4 Burn duration from OAMP and MESA

Target perigee (km)	Predicted duration (sec)	Measured duration—MESA (sec)	% difference Predicted—MESA
150	120	115.00	+4.3
145	49	47.75	+2.6
150	52	48.75	+6.7
145	63	61.75	+2.0
140	47	46.25	+6.1
145	53	51.00	+3.9
140	–	–	–
148	83	79.50	+4.4

represent the drag acceleration in conjunction with a simplified model of other perturbing accelerations such as earch oblateness. Algorithms could be developed for real-time onboard updating of satellite ephemerides.

The parameter estimation function will be evaluated to improve short-term predictions (on the order of one week) by enhancing critical parameters in the model with actual measurement data. Prediction improvement in each instance will be evaluated by comparing the results with satellite ephemerides produced in the absence of any accelerometer data.

Finally, potential applications of those data exist in the calibration of "dynamic model compensation" type sequential filters. In particular, filters which have been designed to recover modeling errors in drag can be evaluated with the use of actual drag accelerations derived from these data.

Conclusions

Accelerometer data received from spacecraft has proved to be an extremely useful tool for real-time maneuver verification in addition to the scientific information content of the data in the study of atmospheric dynamics. The maneuver verification on the Atmosphere Explorer-C satellite was accomplished within 3 min using accelerometer data as compared to greater than 15 min via radio tracking data. In using accelerometer data to evaluate the accuracy of existing atmospheric density models, large deviations are seen between the accelerations predicted by model atmospheres and the measured accelerations.

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