Mission Design Implications of an Inclined Elliptical Geosynchronous Orbit (International Ultraviolet Explorer)

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The U.S./European International Ultraviolet Explorer satellite originally was conceived to consist of a small astronomical satellite placed in an inclined circular geosynchronous orbit over the Atlantic Ocean. However, in order to maximize the net on-station weight capability, a decision was made to alter the original design orbit in favor of an inclined elliptical geosynchronous orbit. This decision provided a unique mission analysis problem and a challenge to meet all mission constraints. This paper discusses the rationale of the elliptical orbit and the resulting mission design implications. The areas of mission analysis included in the paper are 1) trajectory design, 2) launch window analysis, and 3) orbit evolution and stationkeeping analysis.

Introduction

THE International Ultraviolet Explorer (IUE) project is a cooperative program between NASA, the United Kingdom (UK), and the European Space Agency (ESA). The IUE is a small astronomical satellite designed primarily for observing the ultraviolet spectra of stars, galaxies, and the brighter gaseous and planetary nebulae. Figure 1 illustrates the in-orbit configuration of the IUE spacecraft. The project is to place a spectrogaph system in geosynchronous orbit over the Atlantic Ocean.[1] Astronomers at ground observatories then will be able to use the instrument for observations. The IUE observatory consisting of the scientific instrument, spacecraft, and ground system is intended to be used by guest astronomers in a real-time fashion. The system is designed to resemble functionally the operation of a conventional ground-based observatory. This design should limit the amount of special training and maximize the usefulness of the observatory to the astronomical facilities. In addition, it will provide useful experience with a combined satellite/ground-based real-time observatory. NASA will build and operate the U.S. ground observatory at the Goddard Space Flight Center (GSFC). ESA will build and operate the European ground observatory to be located at Villa-franca del Castillo, Spain (VILFRA). The satellite will be inserted into an orbit such that it will be visible to the U.S. ground observatory 24 hr/day and to the European ground observatory a minimum of 10 hr/day. A 3-yr operational lifetime is planned, with a possible 2-yr extension.

The IUE mission initially was conceived to consist of a satellite in an inclined circular geosynchronous orbit. However, in order to increase the net on-station weight (and thus the useful payload), a decision was made to alter the original design in favor of an inclined elliptical geosynchronous orbit. The two candidate missions were studied by simulation using a Monte Carlo program designed for geosynchronous missions.‡ The advantage afforded by the elliptical orbit was measured in terms of the 99th-percentile on-station weight. (This is interpreted properly as 99% of the sample on-station weight values in the Monte Carlo simulation being greater than the 99th percentile.) The elliptical orbit provided a 23-kg weight advantage over the circular orbit. The increase in on-station weight is due primarily to the decrease in the apogee boost motor (ABM) propellant weight. Figure 2 illustrates the increase in on-station weight and the decrease in ABM propellant as a function of apogee bias.§

The nominal apogee bias for the circular mission was determined to be 364 km (200 n. mi.) The ABM propellant weight for this apogee bias is approximately 265 kg. The apogee bias for the elliptical mission was determined to be 10,000 km, which requires 200 kg of ABM propellant. The higher apogee bias for the elliptical mission is an acceptable configuration, since the need for circularization with the hydrazine system has been eliminated. The total difference in the ABM propellant requirements (65 kg) is not realized in on-station weight, since the higher apogee bias also results in a lower allowable liftoff weight (approximately 42 kg) for the Delta 2914 launch vehicle. The 23-kg advantage was sufficient to justify the selection of the elliptical mission orbit.

Nominal Trajectory Design

Although the elliptical mission improves the payload weight, it also poses a challenge in the design of the intermediate and mission orbits. These orbits are subject to constraints dictated by science, thermal, power, communication, and other requirements (Table 1). This section addresses the mission design implications of the elliptical missions and discusses the development of the orbit sequence that satisfies the mission requirements.

Nominal Orbital Sequence

The IUE satellite will be launched by a three-stage Delta 2914 vehicle from the Eastern Test Range at a launch azimuth of 95°. At an altitude of 100 n.m., it will be injected into a circular parking orbit with an inclination of 28.7°. After coasting for the desired duration, the booster will inject the spacecraft into a transfer orbit with a predetermined value of argument of perigee. Following injection into the transfer

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‡The Monte Carlo Investigation of Trajectory Operations and Requirements (MONITOR) Program accounts for errors in the launch vehicle and ABM performance.

§Apogee bias is defined as the height of apogee relative to the synchronous semimajor axis value, 42,164.19 km.

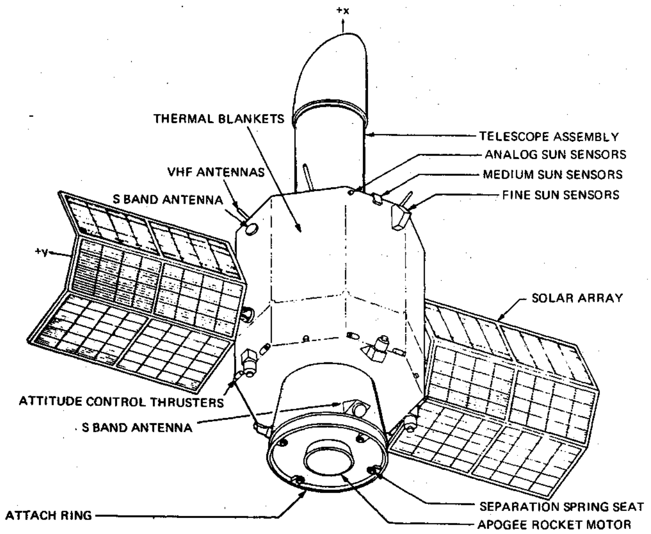

Fig. 1 In-orbit configuration of the IUE spacecraft.

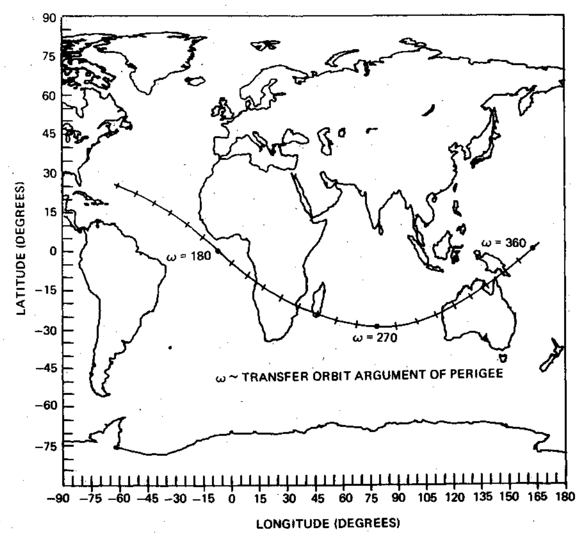

Fig. 3 Parking orbit groundtrack.

orbit, the spacecraft orbit and attitude will be computed from tracking and telemetry data. The hydrazine reaction control system then will reorient the spacecraft into the apogee motor firing attitude. On the second apogee passage of the transfer orbit, the apogee boost motor (ABM) will inject the satellite into a near-synchronous drifting orbit. The ABM will perform a plane change maneuver if required, resulting in the desired inclination and node. The longitudinal drift rate will be nulled by the hydrazine system, resulting in the final elliptical synchronous orbit. In order to satisfy the satellite visibility requirements for GSFC and VILFRA, it will be necessary to perform periodic stationkeeping maneuvers throughout the mission. The maneuvers will be to control the geographic longitude of the ascending nodal crossing.

Mission Requirements and Contraints

A large number of constraints and requirements have been imposed on the IUE mission by the selection of the launch vehicle, spacecraft systems, and scientific goals.[2] The list presented in Table 1 represents the 24 constraints and requirements that affect in some way the orbital selection process. These are listed by mission phase with the source of the requirement and the effect on the mission design. The mission design must account for each of the listed requirements. However, the list can be reduced further to eight items that dominate the orbital design process. These requirements are emphasized in the section dealing with the trajectory parameter selection:

1) The Delta 2914 launch vehicle with a Thiokol 604 apogee engine will be used. The implications of this requirement have been discussed previously.

2) The scientific payload weight capability is to be maximized. This is achieved by minimizing the total of the ABM and hydrazine requirements.

3) The synchronous orbit eccentricity must be less than 0.3 to avoid radiation and communication interference.

4) The synchronous orbit perigee is constrained to be at a southernmost latitude for the mission duration. This is to avoid radiation problems due to the Van Allen radiation belt.

5) The satellite must be visible from GSFC 24 hr/day. This requirement is imposed for both the prime and backup antennas with the actual composite masks.

6) The satellite must be visible from VILFRA at least 10 hr/day above the physical mask plus 10°.

7) The extended mission lifetime is to be 5 yr.

8) The nominal ABM burn will take place near the second apogee of the transfer orbit. This is the earliest reasonable opportunity for the burn. (A first apogee burn is considered feasible only in case of a contingency situation.)

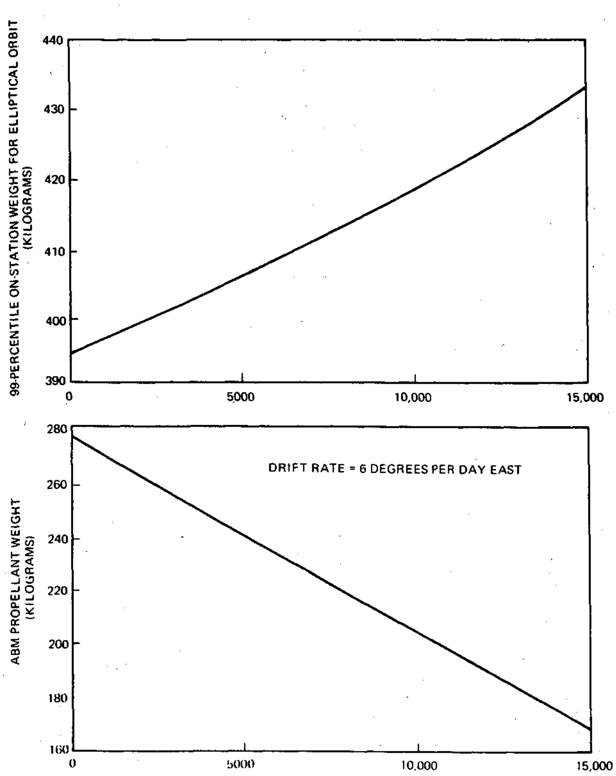

Fig. 2 Weight advantage as a function of apogee bias.

Trajectory Paremeter Selection

The problem of designing the IUE mission, from an orbital standpoint, is essentially one of determining the orbital elements of the transfer and mission orbits.[3] This is an iterative process because of the multiplicity of constraints that must be satisfied. The nominal orbits that were selected are summarized in Table 2.

The orbital elements in Table 2 are based on a spacecraft weight at apogee motor firing (AMF) of 1391 lb (631 kg). The parking orbit elements are typical for a synchronous mission using the Delta 2914 launch vehicle. The node of the parking orbit is related linearly to the launch time and is specified by the launch window. This will be examined in more detail in a later section.

The transfer orbit design reduces to the selection of an apogee bias and argument of perigee. The perigee radius is the same as the parking orbit radius. A change in the orbital plane

Table 1 Mission requirements and constraints

Requirement or Constraint	Value	Source	Effect on Mission Design
A. LAUNCH PHASE			
1. Launch vehicle	Delta-2914	Delta	Limits on injection weight and apogee bias
2. Launch date	Nominal: July 1977	Project	Epoch date, launch window
3. Launch azimuth	Nominal: 95 degrees	Range safety	Parking orbit and transfer orbit elements, tracking coverage
4. Launch window duration	Minimum of 20 minutes	Delta, Project	Launch date and parking orbit node (Ω)
B. PARKING ORBIT PHASE			
5. Altitude	90-100 nautical miles	Payload weight, thermal	Parking orbit and transfer orbit elements
6. Coast time	30 ± 10 minutes	Delta	Transfer orbit elements
7. Injection point	After the descending node equator crossing	Delta	Transfer orbit Argument of perigee (ω)
C. TRANSFER ORBIT PHASE			
8. Duration	Less than 50 hours	Thermal, nutation control	Nutation fuel consumption, ABM firing apogee
9. Perigee height	Minimum: 90 nautical miles and attitude spin axis reorientation must not lower perigee height	Radiation, thermal	Transfer orbit elements
10. Shadow duration	Maximum of 60 minutes per revolution	Thermal, power	Launch window
11. Solar aspect angle	Between 45 and 135 degrees	Thermal, power	Launch window
12. Sun-earth separation angle	Between 30 and 150 degrees at apogee and for the 3 hour interval prior to apogee	Attitude determination	Launch window
D. ABM FIRING AND DRIFT ORBIT PHASE			
13. On-Station arrival time	Within 30 days Nominal: 7 to 14 days	Scientific	Drift rate, apogee for ABM firing
14. ABM firing apogee	Apogee 1 to 4 Nominal: second apogee	Nutation control On-station arrival time, tracking coverage, operational timeline	ABM burn, tracking coverage, fuel consumption, operational timeline, orbital element
15. Ground station coverage	The last 4 hours prior to the apogee burn	Attitude and orbit determination	ABM burn apogee
16. Apogee motor	Thiokol 604-1: 489 lbs. of fuel and inert consumables, 280.6 seconds specific impulse	Project	Payload weight, orbital element
17. Apogee motor pointing error at ABM burn	Within 5 degrees	Attitude	Synchronous orbital element and dispersions
E. SYNCHRONOUS ORBIT PHASE			
18. Period	Synchronous (23.93 hours)	Scientific	Semimajor axis
19. Mission controlled orbit lifetime	5 Years	Project	Stationkeeping maneuvers, fuel consumption
20. Scientific payload weight	Maximize	Project, scientific	Orbital elements
21. Synchronous orbit eccentricity	Less than 0.30	Radiation, communication	Orbital elements
22. US ground station coverage	24 hours per day (with S-Band and VHF) from Greenbelt, MD above 12 m Antenna composite mask	Ground systems, project, networks	Orbital elements, stationkeeping maneuvers
23. European ground station coverage	10-12 hours per day from Vilfra, Spain above physical mask plus 10 degrees	Ground systems, project, ESA	Orbital elements, stationkeeping maneuvers
24. Synchronous orbit shadow duration	Less than 72 minutes	Thermal, scientific	Orbital elements

(inclination and node) was avoided in order that the total energy of the Delta third stage could be used for in-plane adjustment. Thus, the spacecraft weight in the transfer orbit would be maximized. The argument of perigee was selected on the basis of satisfying the radiation and satellite visibility constraints in the synchronous orbit.

The argument of perigee for both transfer and synchronous orbits is determined by the coast time in the parking orbit. An argument of perigee of 257° corresponds to a coast time of 34 min. Figure 3 illustrates the parking orbit groundtrack and the corresponding transfer orbit argument of perigee values at selected injection points. The selection of the apogee bias was subject to three distinct design considerations:

1) The weight of the ABM propellant and inert consumables must be between 400 and 489.1 lb (200 and 220 kg). This is the

Table 2 Nominal orbits

Parameter	Parking orbit	Transfer orbit	Synchronous orbit
Semimajor axis, km	6563.	29,642.	42,164.
Eccentricity	0.0	0.7786	0.2503
Inclination, deg	28.7	28.7	28.7
Node, deg	180.	180.	180.
Argument of perigee, deg	...	257.	257.
Perigee radius, km	6563.	6563.	31,608
Apogee radius, km	6563.	52,720.	52,720.

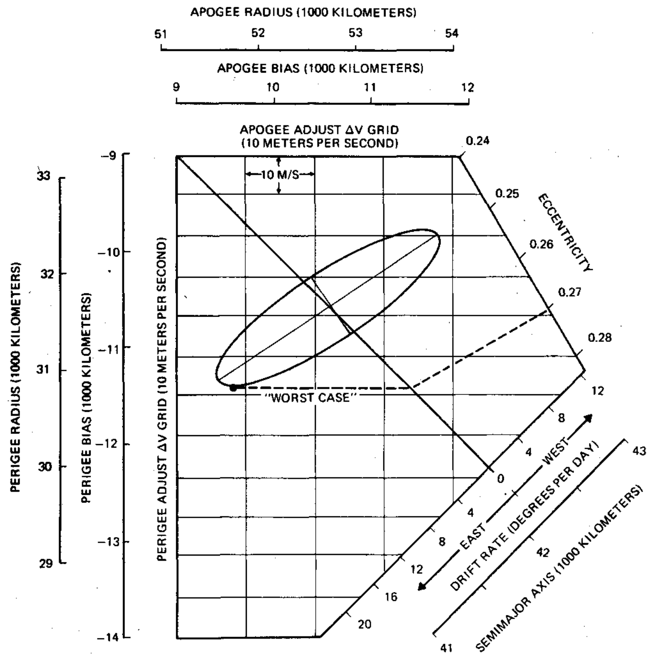

Fig. 4 0.99-probability dispersion ellipse for apogee and perigee radius.

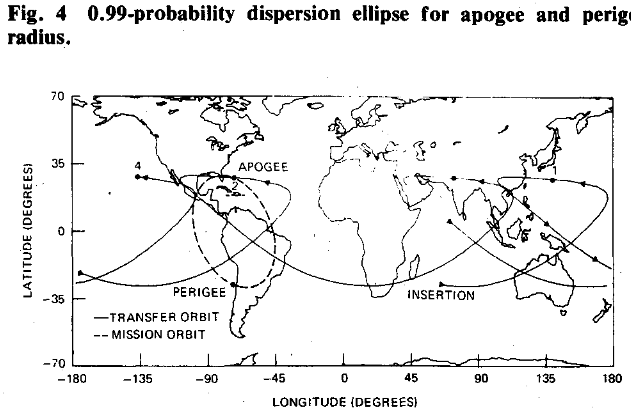

Fig. 5 Transfer and synchronous orbit groundtracks.

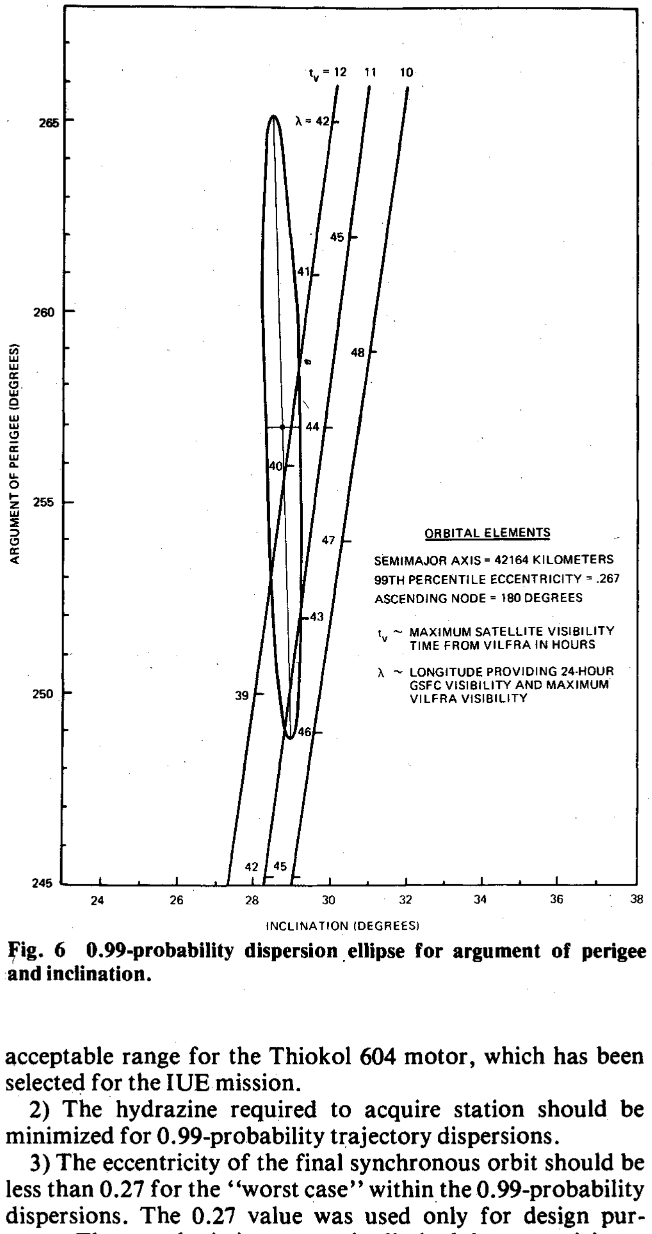

Fig. 6 0.99-probability dispersion ellipse for argument of perigee and inclination.

acceptable range for the Thiokol 604 motor, which has been selected for the IUE mission.

2) The hydrazine required to acquire station should be minimized for 0.99-probability trajectory dispersions.

3) The eccentricity of the final synchronous orbit should be less than 0.27 for the "worst case" within the 0.99-probability dispersions. The 0.27 value was used only for design purposes. The actual mission constraint limited the eccentricity to be less than 0.3. However, smaller eccentricities are preferred from both radiation and tracking standpoints.

Figure 4 presents the 0.99-probability dispersion ellipse in terms of apogee and perigee radius after the ABM burn. The nominal apogee bias of 10,556 km requires an ABM propellant weight of 454 lb (206 kg) for a 0-deg/day drift rate.¶ This drift rate guarantees a minimum station acquisition hydrazine budget considering all points within the dispersion ellipse. Superimposed over Fig. 4 is a relative apogee/perigee adjust grid, each unit being 10 m/sec (4.1 lb of hydrazine). Since the drift orbit is elliptical, it requires more fuel to raise perigee than to lower apogee for a corresponding change in semimajor axis.

To achieve on-station conditions, the satellite must be maneuvered to an acceptable station longitude (geographic longitude of the ascending equatorial crossing) and the drift rate nulled. The procedure to be used for IUE requires use of the hydrazine system to control the drift rate. Since perigee burns are more efficient, it reasonably can be assumed that

¶The term "drift rate" refers to the longitudinal motion of the satellite equatorial crossing in Earth-fixed coodinates. For orbits with orbital periods less than 24 hr, the motion would be eastward. A 0-deg/day drift rate would imply a synchronous orbital period. Westward drift would result from orbits with periods greater than 24 hr.

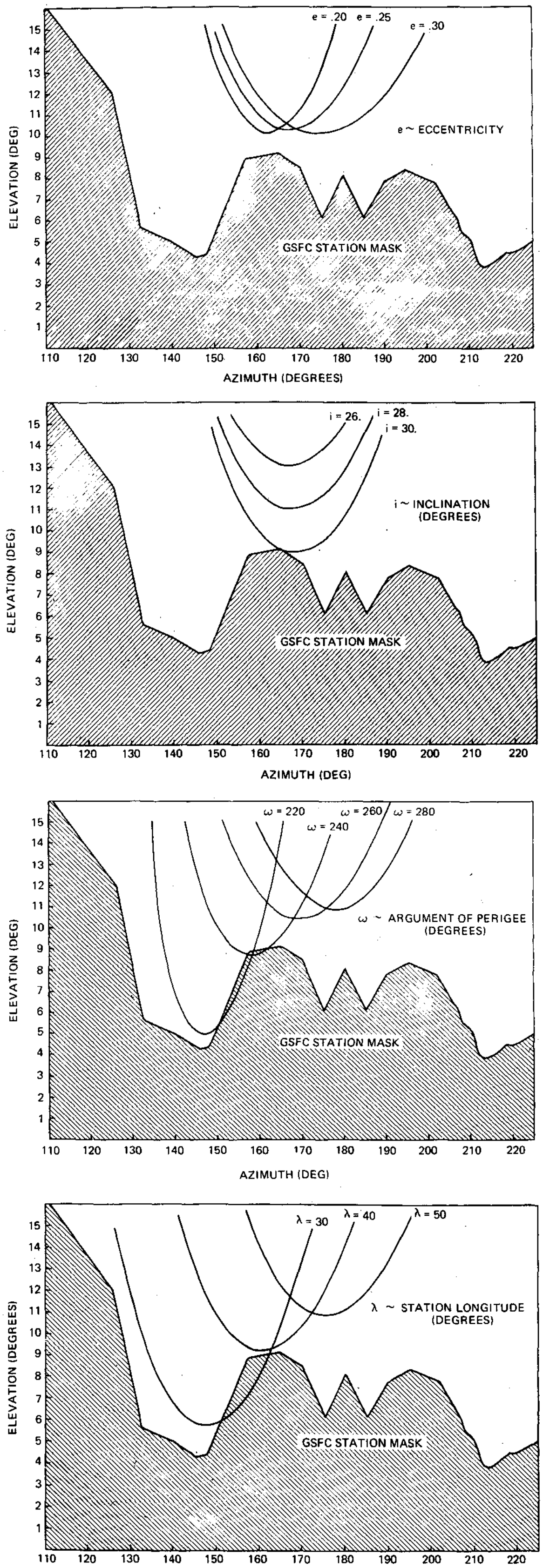

Fig. 7 Effects of parameter variation on GSFC visibility.

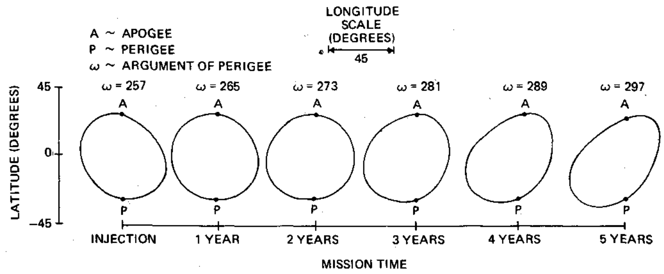

Fig. 8 Synchronous orbit groundtrack evolution.

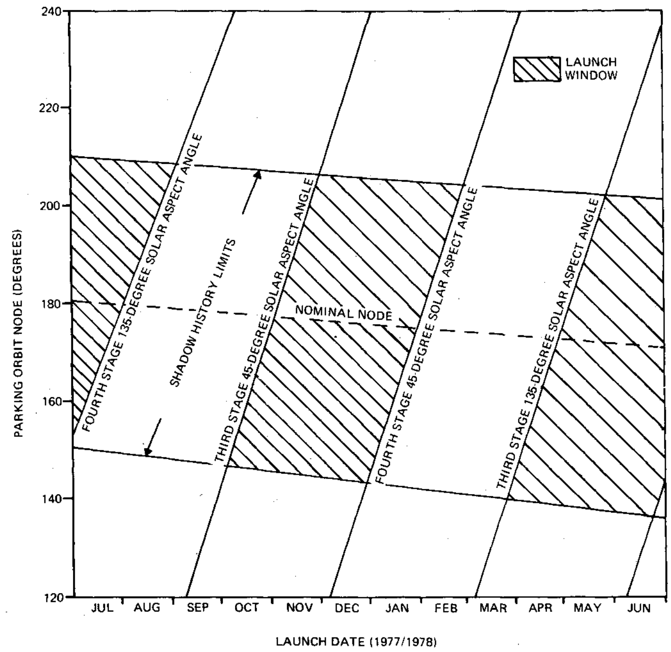

Fig. 9 IUE launch window.

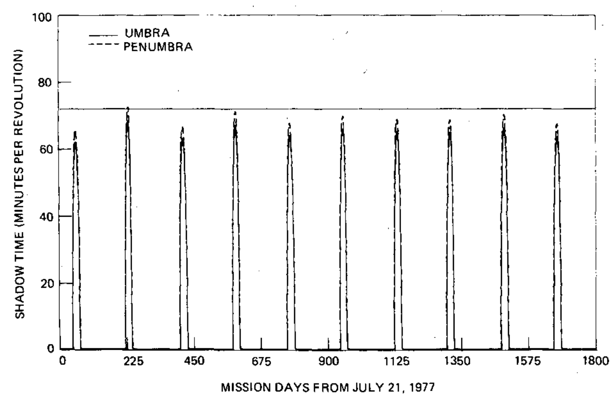

Fig. 10 Synchronous orbit shadow history for a July 21 launch.

only the apogee radius will be adjusted. Using this procedure would make the "worst-case" maximum synchronous orbit eccentricity approximately 0.27, as indicated in Fig. 4. The center of the ellipse is the target nominal and corresponds to an eccentricity of 0.25 and a zero drift rate (semimajor axis = 42,164 km).

Figure 5 gives the ground track of the nominal transfer orbit from insertion to fourth apogee passage and the initial synchronous orbit. The nominal profile has AMF occurring at second apogee of the transfer orbit. The range of acceptable station longitudes for the nominal mission (based on ground station visibility requirements) is 40° to 48° long W. The initial nominal value is 43° (Fig. 5), and the nominal orbit after AMF is synchronous so that no station acquisition maneuvers will be necessary if the target conditions are achieved.

Satellite visibility in the synchronous orbit is controlled by inclination, argument of perigee, and eccentricity. Figure 6 presents the 0.99-probability ellipse in inclination and argument of perigee space. The 99th percentile high value of

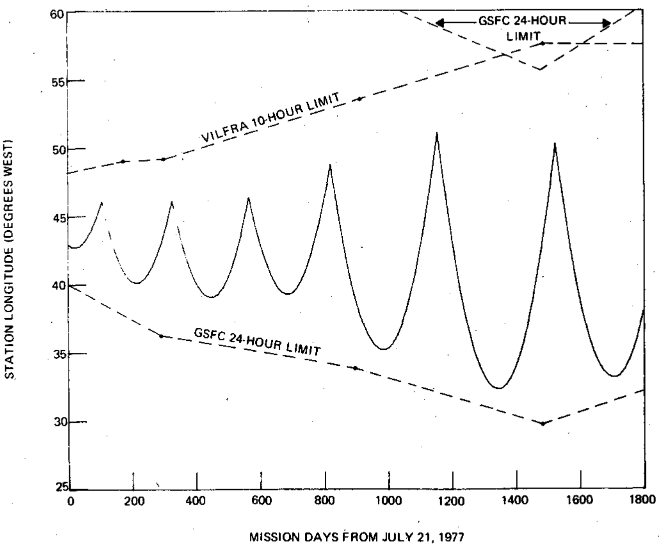

Fig. 11 Evolution of station longitude with stationkeeping.

eccentricity ("worst case") was assumed for the plot (0.267). The nominal inclination (28.7°) and argument of perigee (257°) places the ellipse so that the 24-hr visibility requirement from GSFC visibility and maximum VILFRA visibility. Figure 7 illustrates the effects that inclination, argument of perigee, eccentricity, and station longitude have on GSFC satellite visibility.

Argument of perigee of 257° also satisfies the synchronous orbit radiation constraint by placing perigee at the southernmost latitude for the mission duration. Figure 8 illustrates the 5-yr evolution of the argument of perigee and the corresponding synchronous orbit groundtrack. At 270°, the groundtrack is approximately circular, and perigee is at 28.7° S lat. This occurs near midway of the proposed 5-yr extended lifetime. The node value of 180° was selected to satisfy the shadow constraints for the mission lifetime. This is a launch window constraint and will be discussed in more detail in the next section.

Launch Window Analysis

Constraints

The launch window presents the acceptable launch opportunities in terms of parking orbit node and launch date. The problem of determining the most suitable launch opportunities for a given mission requires consideration of many factors. Six distinct launch window constraints must be satisfied in order to have an acceptable opportunity:

1) Maximum continuous shadow duration prior to apogee motor fire (AMF) must be less than 60 min.

2) Third-stage solar aspect angle must lie between 45° and 135°.

3) Fourth-stage solar aspect angle must lie between 45° abd 135°.

4) Earth-spacecraft-sun separation angle must be between 30° and 150° for the 3-hr period prior to and at apogee.

5) Minimum daily launch window duration must be at least 20 min.

6) Synchronous orbit shadow must be less than 72 min/rev for the entire mission (5 yr).

The pre-AMF shadow constraint must be satisfied from fairing injection up to AMF. The solar aspect angle is measured between the spacecraft spin axis and the spacecraft-sun vector. The separation angle is measured between the spacecraft-sun vector and the spacecraft-Earth vector. The third stage injects the satellite into the transfer orbit from the parking orbit. The fourth stage is the apogee burn motor (ABM) and is used to inject the satellite into the desired drift orbit.

Launch Window Determination

The principal tool used in the parametric launch window analysis was for Fortran Launch Analysis Program (FLAP). The IUE launch window analysis consisted of three basic steps:

1) Independent parameters input to the FLAP were defined by relating them to the IUE mission profile.

2) Mission constraint parameters and their range of values were specified for appropriate constraint plots.

3) Individually generated mission constraint plots were superimposed to form an overall composite plot reflecting time intervals in which no launch time constraints were violated.

The final composite launch window for a 1-yr period is presented in Fig. 9. The ordinate for the plot is right ascension of the ascending node Ω. The absissa of the plot is launch date, which ranges for 1 yr from July 1977. The launch window is closed completely for the months of September and March.

The constraint having the greatest impact on the trajectory design is the synchronous orbit shadow restriction. The maximum allowable continuous shadow for this orbit is 72 min. Because of the eccentricity of the synchronous orbit, shadow durations can be as long as 95 min. The shadow duration is controlled by selection of the initial node and argument of perigee. This constraint essentially limits the range of acceptable node values and hence the daily launch window duration. Figure 10 presents the synchronous orbit shadow history for the nominal initial node of 180°. The nominal initial node of 180° with the nominal initial argument of perigee of 257° proves to be an acceptable combination in limiting the continuous synchronous orbit shadow to 71 min or less for 5 yr.

Stationkeeping and Orbit Evolution

After the spacecraft is placed into an elliptical synchronous orbit, the telescope will be used for observations. The spectrograph image data then will be transmitted to the ground observatories (GSFC and VILFRA). The actual satellite visibility time from the two locations depends on the relative position of the orbit with respect to the sites. In order to satisfy the 24-hr GSFC and 10-hr VILFRA visibility requirements, the evolution of the orbit must be controlled by periodic stationkeeping maneuvers. These maneuvers are designed to control the station longitude by adjusting the semimajor axis. Since perigee burns are more economical than apogee burns, only the apogee radius will be adjusted. Figure 11 presents the evolution of the station longitude with a

Table 3 Maneuver sequence

Mission time days	Perigee radius, km	Initial apogee radius, km	Final apogee radius, km	ΔV, m/sec
105	31,535.36	52,808.43	52,779.04	−0.41
325	31,409.82	52,938.27	52,902.78	−0.50
565	31,335.40	52,994.19	52,957.20	−0.52
820	31,279.71	53,071.83	53,025.49	−0.65
1155	31,247.10	53,110.70	53,053.30	−0.80
1525	31,252.32	53,107.49	53,048.89	−0.82

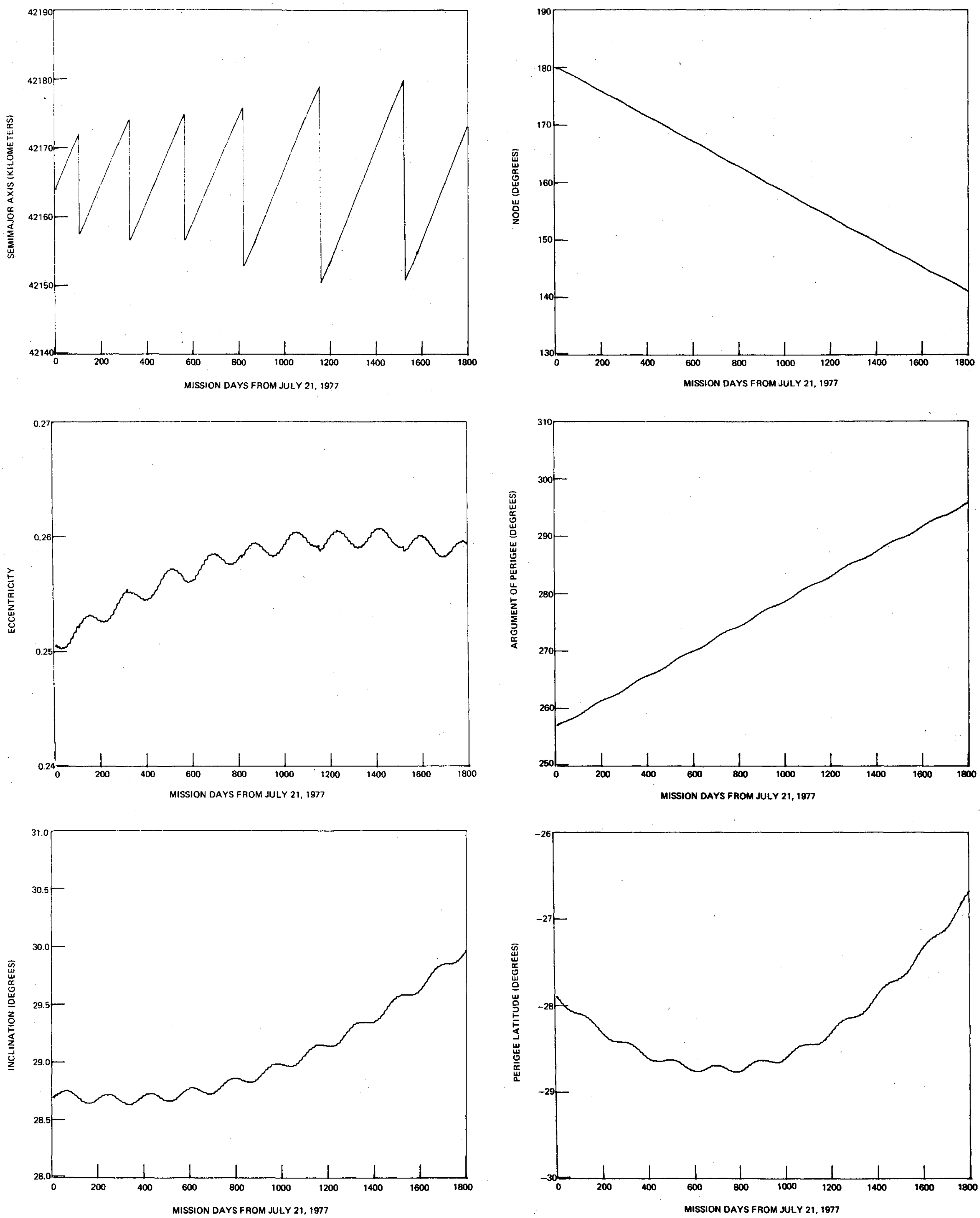

Fig. 12 Evolution of orbital elements with stationkeeping.

typical sequence of stationkeeping maneuvers. The stationkeeping limits also are shown on the figure. The data were generated for 1800 days, including the effects of the moon, sun, and a fourth-order Earth potential model. The particular maneuver sequence that was used is summarized in Table 3.

The maneuvers were designed to maintain the station longitude within the stationkeeping limits with a minimum number of maneuvers. The total ΔV requirement is 3.7 m/sec. This corresponds to 1.5 lb of hydrazine, assuming an on-station weight of 930 lb and a specific impulse of 230 sec. This corresponds to 1.5 lb of hydrazine, assuming an on-station weight of 930 lb and a specific impulse of 230 sec. Increasing the number of maneuvers does not affect the fuel requirements significantly.[4] The corresponding evolution of the orbital elements is presented in Fig. 12. The evolution of the latitude of perigee also is shown, the minimum occurring after approximately 2 yr. Only semimajor axis and eccentricity are affected by the stationkeeping maneuvers.

Summary

The decision to place the IUE satellite into an inclined elliptical geosynchronous orbit provided a challenge to the mission design team to satisfy the multiplicity of mission requirements and constraints. However, intermediate and mission orbits were designed which satisfied all requirements. A significant result of the analysis is the manner in which statistical data (generated by a Monte Carlo simulation) were incorporated into the preflight mission analysis.

Acknowledgment

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