NASA CR-143661

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 1: ORBIT/LAUNCH VEHICLE TRADEOFF STUDIES AND RECOMMENDATIONS

(NASA-CR-143661) EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY. REPORT NO. 1: ORBIT/LAUNCH VEHICLE TRADEOFF STUDIES AND RECOMMENDATIONS (Grumman Aerospace Corp.) 93 p HC $4.75 CSCL 22B G3/18

N75-15707

Unclas 09244

GRUMMAN

UNITED STATES GOVERNMENT

Memorandum

TO : Virginia Kendall
Code 251.2

DATE: February 3, 1975

FROM : D. M. Witters
EOS Study Office

SUBJECT: Release of Reports Thru the STAR NASA Information System

It is requested that all the reports from the EOS Definition Study be released thru the STAR system. This study was performed by three contractors: General Electric, Grumman Aerospace Corporation, and TRW (Thompson-Ramo-Woolrich). Copies of these reports have been furnished to you. A complete set of reports consists of seven reports from each of the three contractors as noted below:

Contractor	Contract	No. of Volumes
General Electric	NAS 5-20518	13
TRW	NAS 5-20519	9
Grumman Aerospace	NAS 5-20520	15

	Report No.	Volumes
General Electric	1	1
	2	1
	3	1
	4	1
	5	7
	6	1
	7	1
TRW	1	1
	2	1
	3	1 + 2 Bk. App.
	4	1
	5	1
	6	1
	7	1
Grumman Aerospace	1	1
	2	1
	3	1 + 6 Vol. App.
	4	1
	5	2
	6	1
	7	1 + 1 Vol. App.

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY

REPORT NO. 1: ORBIT/LAUNCH VEHICLE TRADEOFF STUDIES AND RECOMMENDATIONS

Prepared For

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER
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BETHPAGE, NEW YORK 11714

NAS 5-20520

September 1974

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SUMMARY

1 -SUMMARY

The recommended orbit for the EOS Land Resources Mission should be sun synchronous, with a minimum altitude having acceptable orbit decay, swath sideslip and ground station coverage. The maximum altitude should result in the selection of a low-cost booster, and be capable of direct Shuttle service. The specific altitude selected should be optimized for instrument swath width and the desired repeat cycle time. Our studies indicate that an altitude range of 365-385 n mi is best suited to these EOS requirements.

A promising sun synchronous orbit for EOS missions A, B and C is 366 n mi (678 km) when using an instrument with a 100-n mi (185-km) swath width. This orbit has a 17-day repeat cycle, and a 14-n mi swath overlap. The adjacent western swath overlap occurs in 3 days; the eastern in 14 days. When using a High Resolution Pointing Imager (HRPI) instrument with 30-deg offset pointing in CONUS viewing, 90% of a reference swath may be viewed again within three days.

For a nine-day repeat cycle time, an acceptable orbit within the recommended altitude is 382 n mi (708 km). This orbit should be operated with an instrument whose swath width is 178 n mi (330 km). It provides a Thematic Mapper (TM) swath overlap of 15 n mi and an adjacent swath overlap will occur in two days.

The 366-n mi orbit was evaluated for orbit decay and was found operationally acceptable; that is, the node sideslip at the equator was ± 2.1 n mi in 30 days and ± 8.5 n mi for a 60-day period (assuming a 1979 nominal atmosphere).

The lower 366-n mi orbit was also evaluated for ground station coverage from Sioux Falls. Since this data acquisition ground station is not yet operational, the site survey data, which predicts a clear field to within two degrees of the horizontal, was used. With this horizon mask, our analysis indicates complete coverage of CONUS for the 366-n mi and higher orbit altitudes.

Payloads were developed for each mission, EOS A through F. A detail weights analysis of the spacecraft design was used in developing payload weights. For each mission, the lowest cost booster that was capable of lifting the payload to the EOS orbit was selected. The Delta 2910 launch vehicle was selected for the A and D missions; the Delta 3910 for the B and E missions; the Titan III (SSB), with a payload-integrated SRM, for the C mission, and the Titan III C7 for the EOS-F mission.

At the 366-n mi operational altitude recommended for Mission A, B and C, direct Shuttle service is possible. The selection of higher altitudes may require the addition of payload-integrated multi-SRM kick stages to transfer the spacecraft to a lower orbit for shuttle service.

All EOS missions would be launched from WTR except for F, which would be an ETR launch. This mission will require a 7-segmented SRM Titan launch vehicle, the Titan III C7. The Titan III C7 has a Transtage as the upper stage which is used to circularize at geosynchronous altitude. For Shuttle servicing, missions E and F will require special provisions.

Because of the poor reliability (0.89) of the low-cost Delta 2910 launch vehicle, identified for Missions A and D, it is recommended that program planning take into account the possibility of a failed launch.

INTRODUCTION

2 - INTRODUCTION

The selection of the EOS orbit and the launch vehicle to be used for payload delivery is the outcome of a number of tradeoff studies involving many system parameters and system costs. To be considered in the selection are such diverse influences as:

- Mapping coverage and swath overlap
- Area revisit intervals
- Lighting at the spacecraft
- Orbit adjust frequency and ΔV needs
- Tracking and data acquisition requirements.

The impact of the foregoing on spacecraft design and operations must also be considered. Additionally, the desirability of utilizing the Shuttle for EOS service and resupply and, ultimately, for initial EOS deploy and retrieve missions, also bears on the orbit altitude selection.

In addition to making minimal cost a principal guide, a number of constraints and assumptions have been imposed on the studies to narrow the wide scope of potential systems that might also be equally satisfactory. Namely,

- The width of the thematic mapper ground swath is taken as 100 n mi (185.2 km). The effect of larger swath widths is considered briefly
- Overlap between adjacent swath widths, referred to as sidelap, should be between 10 and 20 n mi at the equator
- Full earth coverage should be completed in a time period not exceeding that which exists for ERTS (repeat cycle time is 18 days)
- Orbit adjust due to aerodynamic drag, solar radiation pressure and other system external perturbations should be infrequent - preferably no more frequent than every 30 days
- The EOS thermal requirements should be compatible with solar directions associated with a descending node time of day (DNTD) ranging from 9:30 a.m. to 12 noon
- The EOS should operate for a minimum of two years before servicing is required
- Spacecraft operation and data acquisition should be accomplished with STDN ground stations at (1) Fairbanks and Sioux Falls, or (2) Fairbanks, Goldstone, and NTTF for complete CONUS coverage.

This volume provides the results of the various tradeoff studies that lead to a recommendation of the EOS orbit altitude. The tradeoffs which are influenced principally by user needs (e.g., mapping coverage, area revisit, target lighting) are discussed in Section 3, "User Impact on Orbit Selection." Section 4, "Spacecraft Impact on Orbit Selection," contains tradeoff discussions dealing with spacecraft design and how the latter is affected by such considerations as solar lighting and ground communication.

Section 5 presents the mission tracking coverage obtained at the proposed orbit altitude. Descriptions of the various candidate launch vehicles and their "payload-into-orbit" capabilities are included in Section 6. Operational procedures and launch vehicle reliability are also included in Section 6 as are the follow-on missions and their compatibility with the basic EOS design

USER IMPACT ON
ORBIT SELECTION

3 - USER IMPACT ON ORBIT SELECTION

The needs of the user, be it for land resources analysis or cartography, is one of the principal determinants in the design of the EOS system and the orbit selection. This section presents studies on following topics:

- Sun synchronism
- Mapping coverage and its dependence on orbit altitude and repeat cycle time
- Mapping constraints imposed by TM overlap and HRPI offset angles
- Orbit decay and its effect on coverage.

These studies have a major impact in that they are significant drivers toward orbit altitude selection.

Discussions of the impact of solar lighting on user/science applications are presented in Report No. 3.

3.1 SUN SYNCHRONISM

An essential feature of the EOS system and its gathering of earth resources data is the maintenance of essentially the same lighting conditions over a particular area as that area is successively imaged. To achieve this effect, the orbit plane should be selected so as to precess about the earth's spin axis at a rate equal to the earth's mean motion about the sun. This precession, due to zonal harmonic terms in the earth's potential function, depends upon both the orbit inclination and its semi-latus rectum (or radius, in the case of a circular orbit). With the orbit precession rate chosen to be 0.9856 deg/day (earth's mean motion), the dependence of inclination, i, on altitude, h, may be found from $i = 90^{\circ} + 0.0988 \left[\left(\frac{R_e + h}{R_e}\right)\right]^{7/2}$, where R_e is the equatorial radius of the earth. Figure 3-1 shows the relative insensitivity of inclination to altitude change with a change of only 1 deg over the range 300 to 400 n mi (555 to 925 km).

With sun synchronism established by careful execution of final insertion into the desired orbit inclination and altitude, the angle between the orbit node and the sun's right ascension will remain nearly constant during the year. Some periodic variation arises due to the sun's north-south motion associated with the change of seasons. The effect of this motion on the solar panel illumination is treated in Subsection 4.1.

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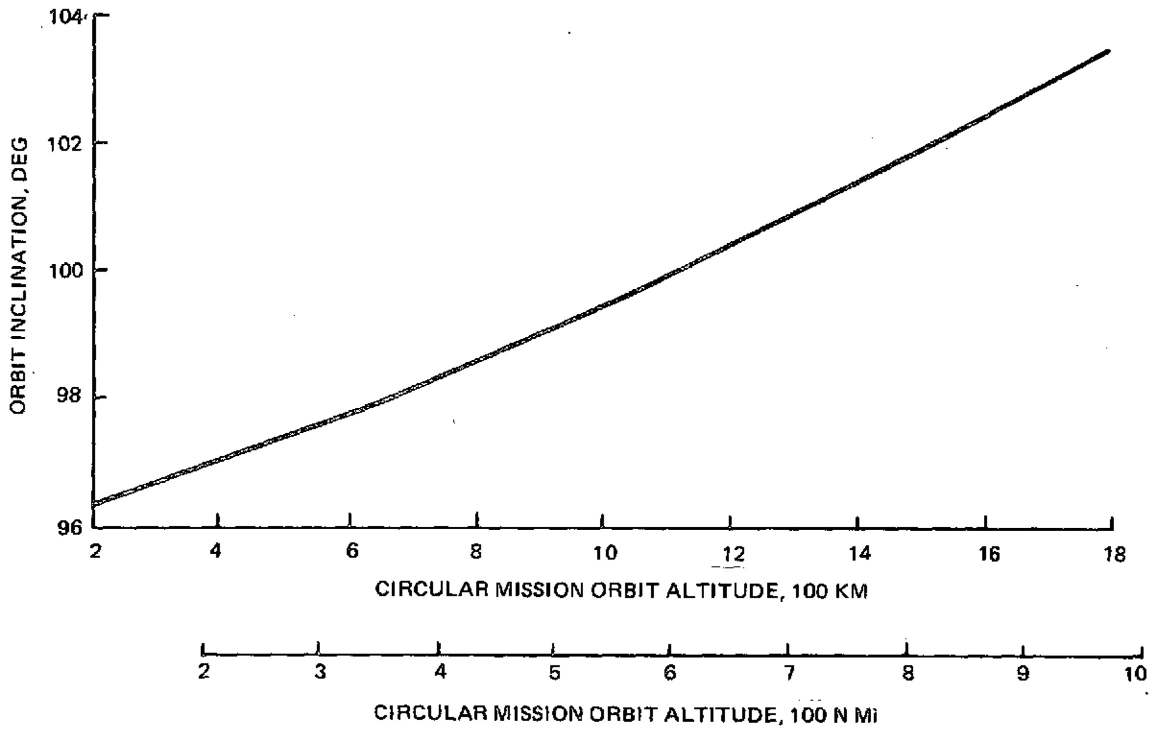

1-1

Fig. 3-1 Sun-Synchronous Orbit Inclinations Versus Altitude

Loss of sun synchronism will result from any perturbations which alter the orbit altitude (and, therefore, the orbit period) or the orbit inclination. The principal source of synchronism degradation is aerodynamic drag. The loss in orbit period per nautical mile drop in altitude is shown in Fig. 3-2. In particular, at 366 n mi (678 km) the period is reduced about 2.32 sec.

It is shown in Subsection 3.3.1 that after three months in orbit the spacecraft will descend approximately 0.1 n mi (0.19 km). At about 14.7 orbits/day (for a 366-n mi altitude orbit and, also, assuming the period loss per orbit to be uniform) there results a 155 sec or 2.6 min loss of in-orbit phasing. This corresponds to a loss in node arrival time such that the earth-relative orbit node will move eastward by 2.6 min x 15 n mi/min = 39 n mi. With corrections made every three months, sun synchronism may be considered as varying by ±1.3 min about a mean descending node time-of-day (DNTD); an insignificant disturbance to sun synchronism. The selection of 366 n mi for the foregoing example will become evident from further studies.

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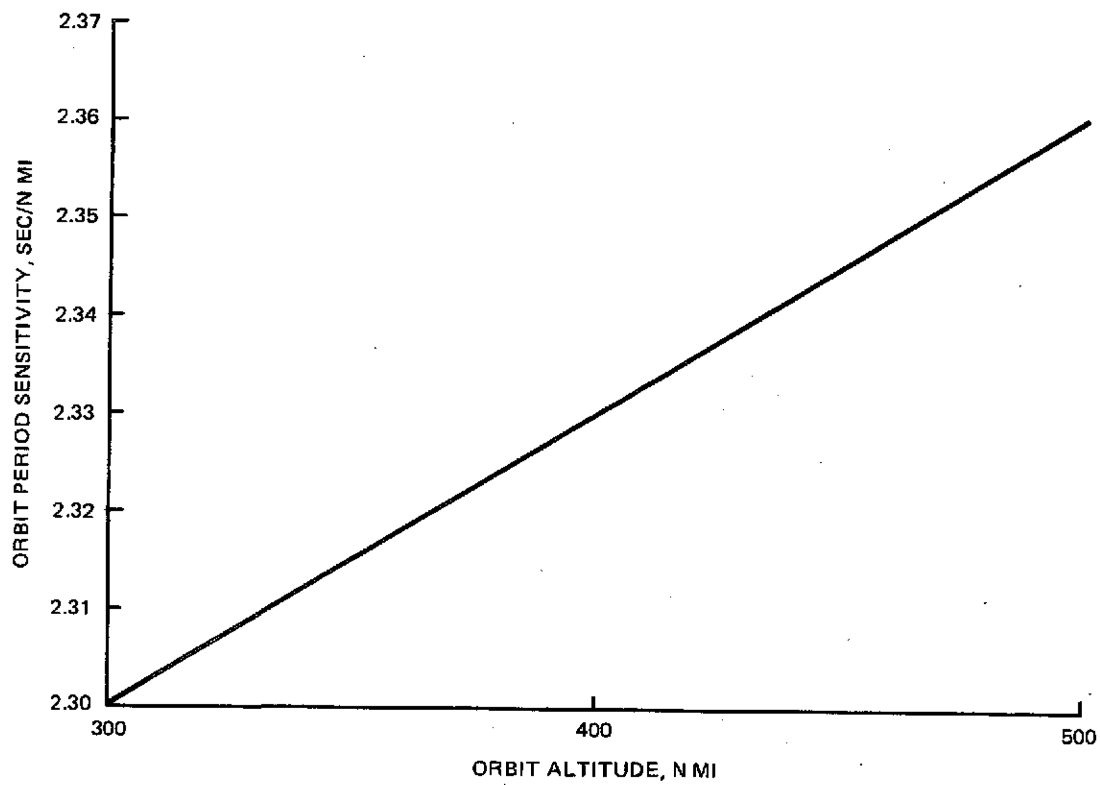

1-2

Fig. 3-2 Orbital Period Sensitivity to Altitude

3.2 MAPPING COVERAGE

The repetitive nature of orbit tracks over the earth's surface is one of the main features of the EOS mission. It allows periodic observation of the same area under the same or very similar lighting conditions. The interval of time during which earth coverage is fully completed by one satellite is referred to as the repeat cycle time; is here designated as N, measured in days. The number of orbits required for complete earth coverage during a repeat cycle is n, an integer. In requiring the initial track of a repeat cycle to be superimposed upon the corresponding initial track of a prior cycle and requiring further that both tracks occur at the same time of day, the value of N is constrained to be an integer. Each combination of integers N and n yields a "solution" which satisfies the periodicity and lighting requirements along with full earth coverage (provided the sensor swath width is adequate for the selected value of n). However, only certain of these "solutions" are realizable in practice since each will be associated with a particular orbit altitude, and most altitudes will fall outside the limits imposed by other considerations (e.g., launch vehicle, experiment instrumentation, tracking, orbit adjust, etc).

3.2.1 EFFECT OF VARYING n

Table 3-1 lists the possible patterns for N=17 and 18 days and for an altitude range of $300 < h < 500$ n mi ($555 < h < 925$ km). For complete earth coverage the sensor swath width m must exceed the separation between adjacent tracks, S. The overlap between adjacent swaths should be the order of 10 to 20 n mi (18.5 to 37 km) to facilitate the matching of adjacent imagery. The term p is the number of days delay until the closest overlapping swath is generated, and m is an index which designates which side (+ 1 for west, - 1 for east) of the swath is overlapped on the p^{th} day. The overlap on the opposite side occurs on the $(N-p)^{th}$ day. The term ΔS is the adjacent swath overlap at the equator in nautical miles based on an assumed 100-n mi (185.2-km) swath width.

The selection of an altitude from among these possible "solutions" is governed by the results of various tradeoff studies. Reducing the effect of air drag, and, therefore, orbit adjust frequency is achieved by raising the orbit altitude. Increased orbit altitude also lengthens the time that the spacecraft remains with Tracking and Data Acquisition (T&DA) antenna coverage. On the other hand, improved imagery resolution, increased physical payload delivery, and Shuttle utilization are fostered by a reduction in orbit mission altitude. In addition to these considerations, benefits derived from the use of the HRPI instrument depend the type of swath lay-down pattern and its impact on the HRPI revisit interval. All of these effects are discussed in later subsections.

To provide a graphical interpretation of the relation between N, n, p and orbit altitude, Fig. 3-3 illustrates the solutions listed in Table 3-1 (including those for N = 16 days, not shown in the table). In general, other repeat cycle times become available by altering the sensor swath width. For example, if the swath width is opened to 178 n mi (330 km) or 262 n mi (485 km), fewer orbits are required to cover the earth. To maintain similar adjacent swath overlap with these swath widths (in the same altitude range), the repeat cycle times should be reduced to 9 and 6 days, respectively. Table 3-2 lists the possible solutions for these repeat cycle times.

Other repeat cycle times down to N = 7 days will yield solutions falling in the altitude range 365 to 385 n mi and each will require a swath width appropriate to the desired adjacent swath overlap. As seen from Table 3-2, N = 6 days does not have a solution in this altitude range. The two solutions shown have the disadvantage of being too low to avoid excessive orbit decay from aerodynamic drag or too high to utilize the Shuttle as a launch vehicle.

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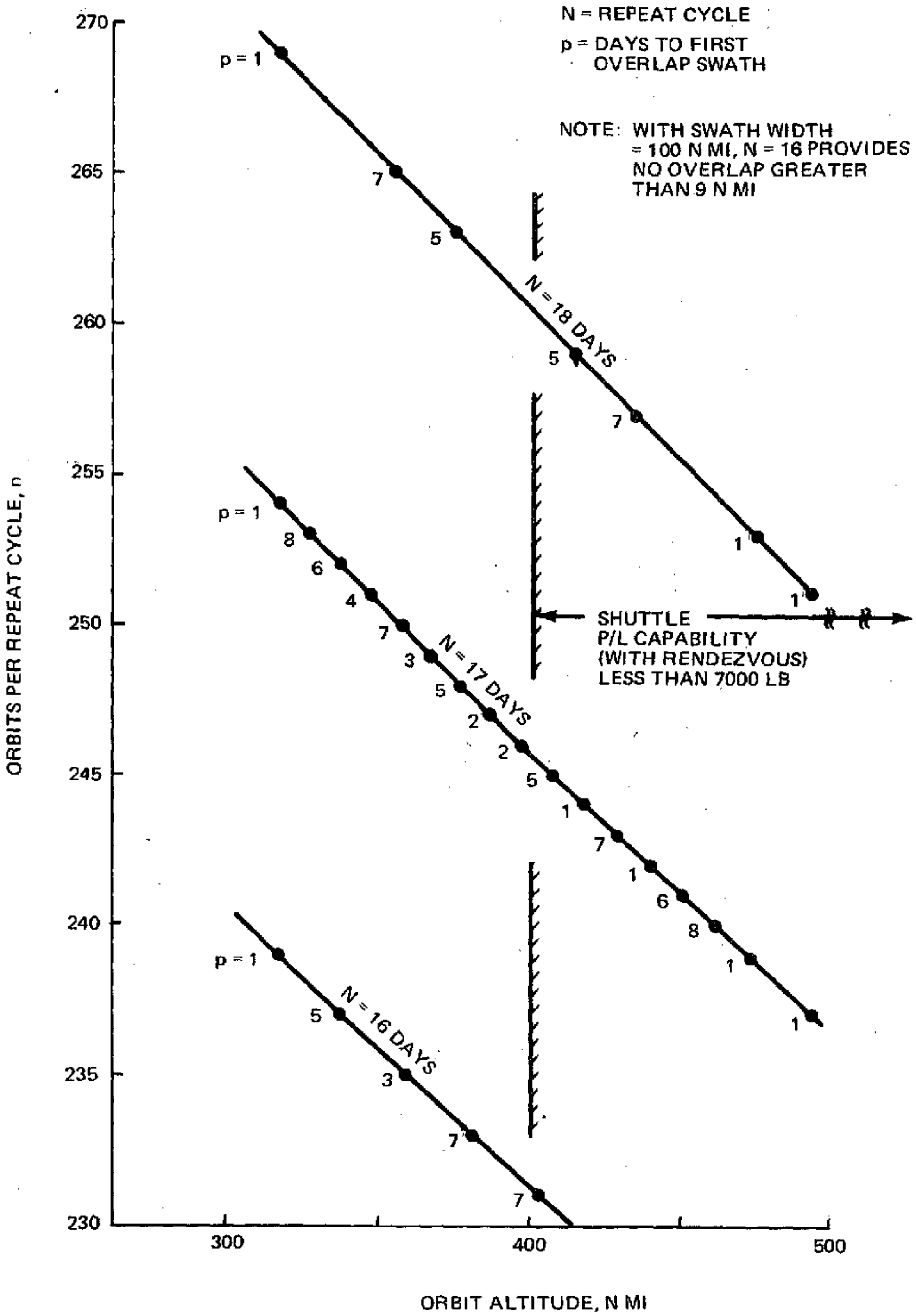

1-3

Fig. 3-3 Candidate Orbit Altitudes Between 300 and 500 Nautical Miles for 16-, 17-, and 18-Day Repeat Cycles

Table 3-1 Possible Swath Patterns for Altitudes Between 300 and 500 Nautical Miles for 17-Day and 18-Day Repeat Cycles

SWATH WIDTH = 100 N MI (185.2 KM)

N = 16 — NO OVERLAP >10 N MI FOR 300 <H <500 N MI

N = 17

n	p	m	h (KM)	h (N MI)	S (KM)	S (N MI)	ΔS (N MI)
237	1	+1	914.2	493.6	169.1	91.3	10.0
	16	-1					
238	1	+1	873.5	417.6	167.7	90.5	10.8
	16	-1					
240	8	+1	853.3	460.8	167.0	90.1	11.2
	9	-1					
241	6	-1	833.3	450.0	166.2	89.7	11.5
	11	+1					
242	4	+1	813.4	439.2	165.5	89.4	11.8
	13	-1					
243	7	-1	793.7	428.5	165.0	89.0	12.2
	10	+1					
244	3	-1	774.1	418.0	164.2	88.6	12.5
	14	+1					
245	5	-1	754.6	407.4	163.6	88.3	12.8
	12	+1					
246	2	+1	735.2	397.0	162.9	88.0	13.1
	15	-1					
247	2	-1	716.0	386.6	162.2	87.6	13.5
	15	+1					
248	5	+1	697.0	376.3	161.7	87.3	13.8
	12	-1					
249	3	+1	678.0	366.1	161.0	86.9	14.2
	14	-1					
250	7	+1	659.2	355.9	160.3	86.6	14.5
	10	-1					
251	4	-1	640.5	345.8	159.7	86.3	14.8
	13	+1					
252	6	+1	621.9	335.8	159.0	85.9	15.1
	11	-1					
253	8	-1	603.4	325.8	158.5	85.6	15.4
	9	+1					
254	1	+1	585.1	315.9	157.8	85.2	15.8
	16	-1					

N = 18

n	p	m	h (KM)	h (N MI)	S (KM)	S (N MI)	ΔS (N MI)
251	1	+1	913.1	493.0	159.7	86.2	15.1
	17	-1					
253	1	-1	874.6	472.2	158.4	85.5	15.8
	17	+1					
257	7	+1	799.1	431.5	155.8	84.2	17.0
	11	-1					
259	5	-1	762.1	411.5	154.7	83.5	17.7
	13	-1					
263	5	-1	689.6	372.3	152.4	82.3	18.7
	13	-1					
265	7	-1	654.0	353.1	151.3	81.7	19.4
	11	+1					
269	1	+1	584.1	315.4	149.0	80.4	20.6
	17	-1					

T1-1

- SWATH WIDTH = 100 N MI
  DUE TO ORBIT INCLINATION THE SWATH COVERS ABOUT A 101.2 N MI SEGMENT ON THE EQUATOR.
- ALL TABLE ENTRIES FOR N = 17 AND N = 18 DAYS ARE RESTRICTED TO 300 <ALTITUDE <500 N MI.

Note that the number of table entries, n, which provide solutions for N = 6 or 9 are significantly less than in the instance of N = 17. This results from 6 and 9 not being prime numbers (as is 17) and, therefore, having common factors with many of the values of n. For example, although n = 88, N = 6 would yield a 365-n mi orbit altitude, the latter is not an acceptable solution since it is equivalent to n = 44 for N = 3; and 44 orbits (per cycle) would not fully cover the earth with a swath width of 262 n mi (485 km).

Table 3-2 Possible Solutions for Orbit Altitude Selection Relative to Repeat Cycle Time, Days Delay, and Swath Overlap

SWATH WIDTH = 178 N MI (330 KM)							
N = 9			h		S		ΔS
n	p	m	(KM)	(N MI)	(KM)	(N MI)	(N MI)
127	1	-1	855.5	462.0	315.5	170.4	10.4
	8	+1					
128	4	+1	817.8	441.6	313.0	169.0	11.0
	5	-1					
130	2	+1	743.8	401.6	308.3	166.4	13.6
	7	-1					
131	2	-1	707.5	382.0	306.0	165.2	14.8
	7	±1					
133	4	-1	636.3	343.6	301.4	162.7	17.3
	5	+1					
134	1	+1	601.4	324.7	299.2	161.6	18.4
	8	-1					
SWATH WIDTH = 262 N MI (485 KM)							
N = 6			h		S		ΔS
n	p	m	(KM)	(N MI)	(KM)	(N MI)	(N MI)
85	1	-1	836.6	451.7	471.5	254.6	10.0
	5	+1					
89	1	+1	618.8	334.1	450.3	243.1	21.5
	5	-1					

T1-2

3.2.2 TWO SATELLITE COVERAGE

The repeat cycle time obtained with the use of one satellite may be effectively halved by placing another, similar satellite in orbit with the same repeat cycle, N, and orbits per cycle, n. Both satellites would be in the same orbit plane and, therefore, have the same DNTD, but they would have an in-plane phase separation that would bring each satellite (alternately) over the same track about N/2 days apart. (If N is odd, full coverage is achieved in (N+1)/2 days. If N is even, it is N/2).

Other phasing could be employed to bring about any desired interval in satellite passage over the same earth track. For example, if meteorological data suggests that cloud formations over any particular area are usually dissipated within three days, the second satellite

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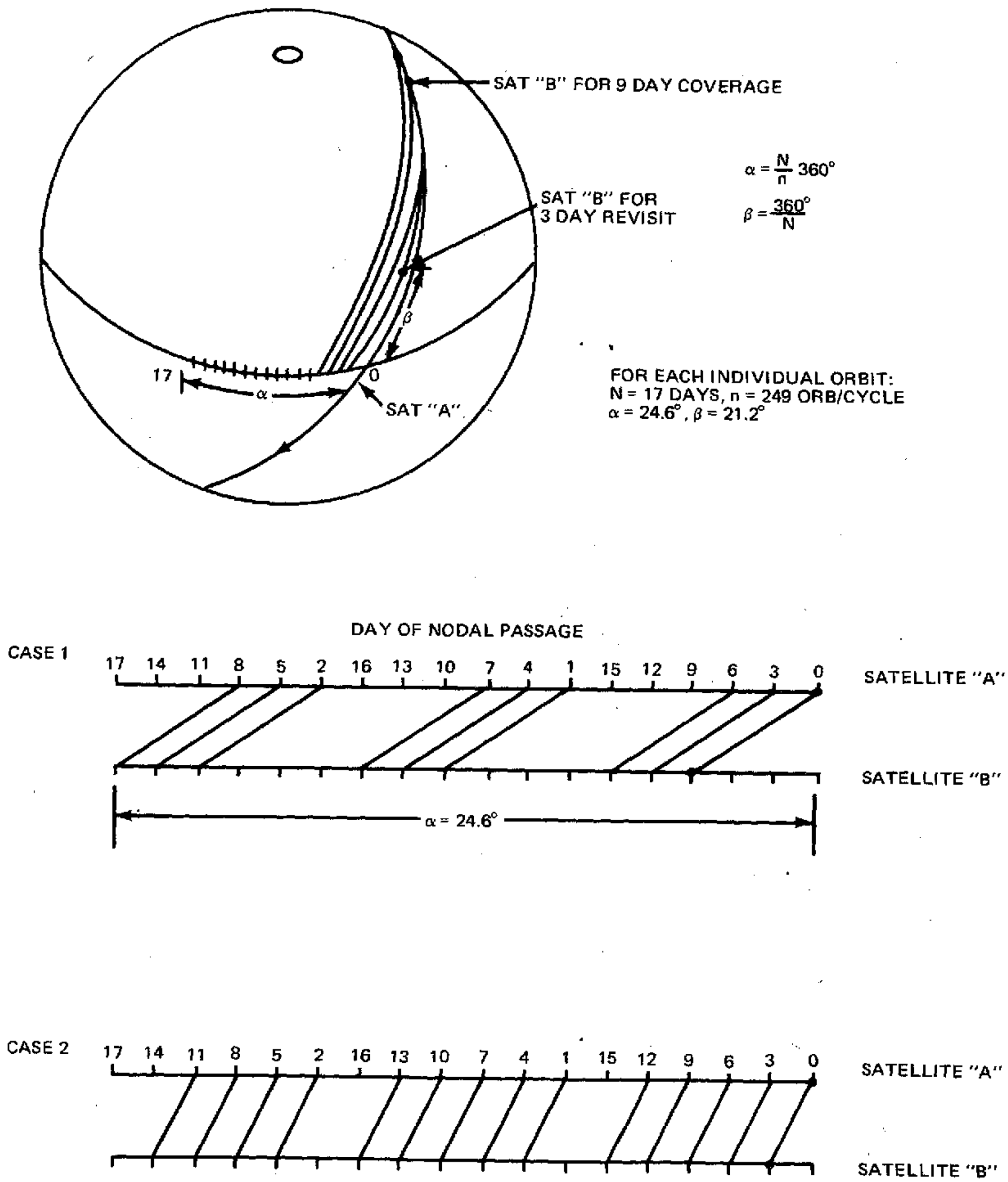
may be phased to cover the same area three days after the first satellite, thereby significantly improving the probability of obtaining cloud-free imagery.

Figure 3-4 is a sketch which depicts some of the ground tracks for satellites simultaneously spaced in the same orbit plane. The equatorial interval, α, is the distance between successive equatorial crossings of one satellite. For a 366.1-n mi (678-km) orbit altitude the segment α is 24.6°. When N=17 days, the successive 17 daily crossings of one satellite mark off the same subdivisions along one α arc (though not in contiguous order) that are generated by crossings of 17 equally spaced satellites during one orbit period. Two cases are given in Fig. 3-4 as examples to demonstrate the geometric relations in a multi-satellite system when N=17 days/cycle and n=249 orbits/cycle for each individual satellite. The swath lay-down pattern is shown as the "day of nodal passage". As shown in Case 1, the 17-day repeat cycle time of one satellite can, with two satellites, be made to provide full earth coverage in 9 days if the satellites are separated in orbit by 62.6 deg. Since both satellites are in the same sun-synchronous orbit, the lighting at each track, of either satellite, is the same.

In Subsection 3.1 it was stated that a 6-day repeat cycle does not provide solutions at an altitude suitable to Shuttle utilization and orbit decay requirements. However, by employing two 12-day satellites whose orbits are in the acceptable 365- to 385-n mi altitude range, an effective 6-day repeat cycle time could be achieved.

Case 2 shows that a 3-day delay between the two satellite passages over the same track can be achieved by spacing the satellites in orbit 21.2 deg apart. The close phasing of the two satellites in this instance, however, being only about 5.8 min apart in orbit, could present a problem in tracking and data acquisition, since ground station coverage of the satellite could last as long as 11 min. This may be overcome by selecting a slightly different sun-synchronous orbit plane, differing by about 6 min eastward in DNTD and delaying the second satellite, backward in its orbit, by an angle equivalent to 6 min of in-orbit travel time.

An obvious extension of the geometry depicted in Fig. 3-4 is that N satellites, equally spaced in the same sun-synchronous orbit plane, with each satellite individually providing an N-day repeat cycle, would, together, provide full earth coverage once each day.



1-4

Fig. 3-4 Ground Tracks for Satellites Simultaneously Placed in Same Orbit Plane

3.2.3 MAPPING CONSTRAINTS

THEMATIC MAPPER - As discussed in Subsection 3.2.1 the selected TM swath width will have a major influence in fixing the number of orbits to be made in a repeat cycle interval. There a 10- to 20-n mi (18.5 to 37 km) adjacent swath overlap was assumed as a design objective. When overlap decreases below 10 n mi the difficulty in matching imagery of adjacent scenes increases. On the other hand, when overlap exceeds about 20 n mi the DMS is needlessly overburdened with redundant data.

HRPI REVISIT - The HRPI is intended for use with ground targets which require more frequent observation than is available through use of the TM alone. Whereas the TM axis is always directed toward the orbit nadir (suborbital point), the HRPI can be laterally offset to some maximum angle on either side of the track. This enables more frequent looks at targets detected by the TM and also provides an earlier detection, with subsequent revisit, to a large number of potential targets not immediately available to the TM.

In selecting an EOS mission altitude, consideration was given to the patterns of coverage provided by HRPI. Of prime interest is the coverage which can be provided within the boundaries of the continental U.S.A. (CONUS). The latitude bounds for CONUS were assumed to be 25°N and 48°N. The TM swath width was assumed to be 100 n mi (185 km) for this investigation.

Figures 3-5 through 3-13 show day-by-day HRPI and TM coverage as a percentage of the targets which were HRPI-accessible on Day 0 (referred to as the reference swath). Note that the 100-n mi TM width, being independent of orbit altitude, becomes a greater percentage of the HRPI accessible swath as the orbit altitude is reduced.

The orbit altitudes presented in these figures were selected by restricting the repeat cycle to 17 and 18 days (implying the assumption of a 100-n mi TM swadth width) and also eliminating those solutions with $p > 4$ days (p is the time to lay down the adjacent overlapping TM swath).

As an example, we note from Fig. 3-5 (orbit altitude 439 n mi) that by Day 3 after an initial HRPI sighting at latitude 25°, 61% of all HRPI accessible targets could have been revisited. This is achieved by having available the eastern 38% on Day 1 and the western 23% on Day 3. The eastern 85% available on Day 4 then assures that 100% of the targets of latitude 25° which could have been seen by HRPI on Day 0 could be revisited at least once more by the fourth day.

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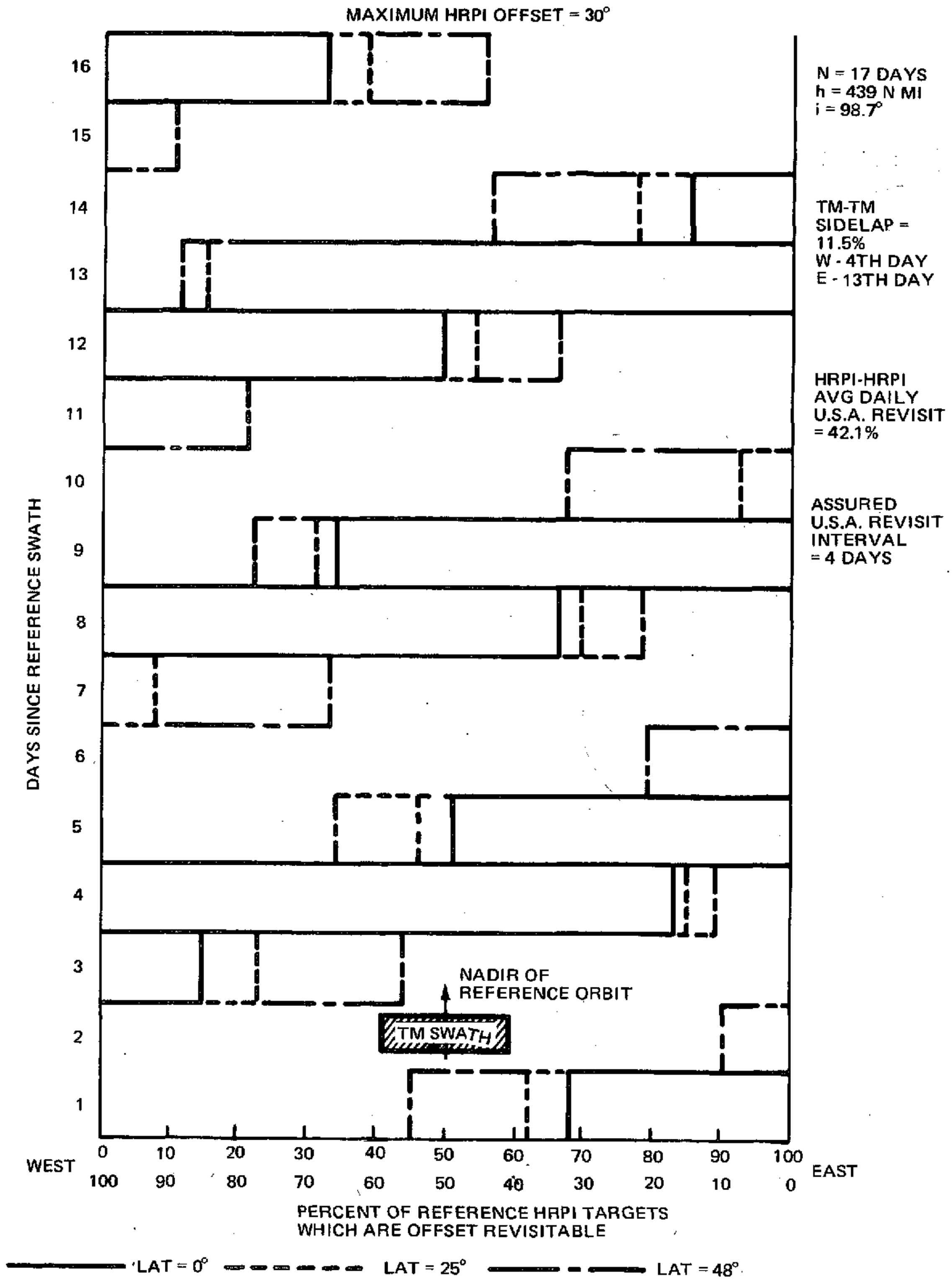
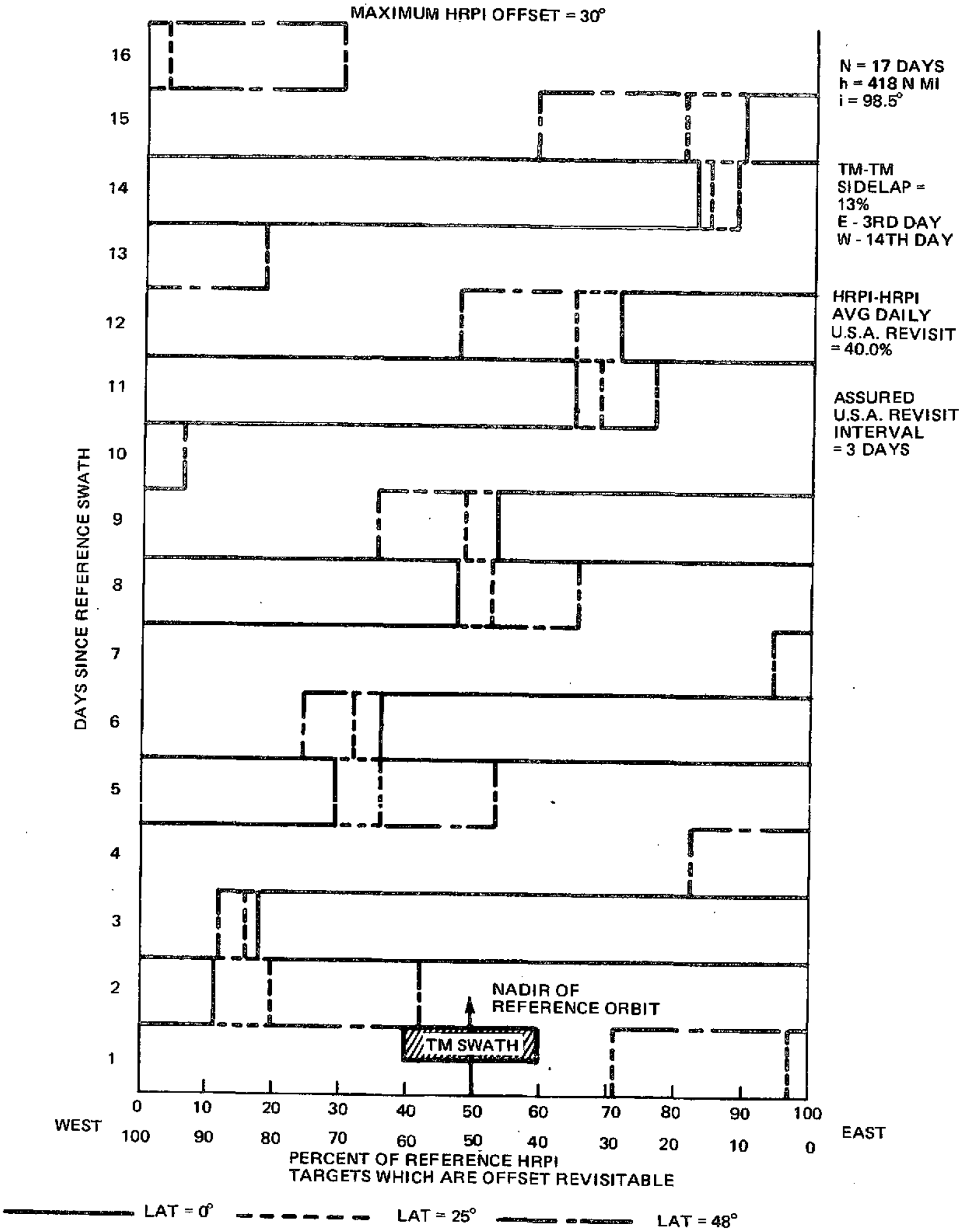


1-5

Fig. 3-5 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 439 N Mi)



1-6 Fig. 3-6 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 418 N Mi)

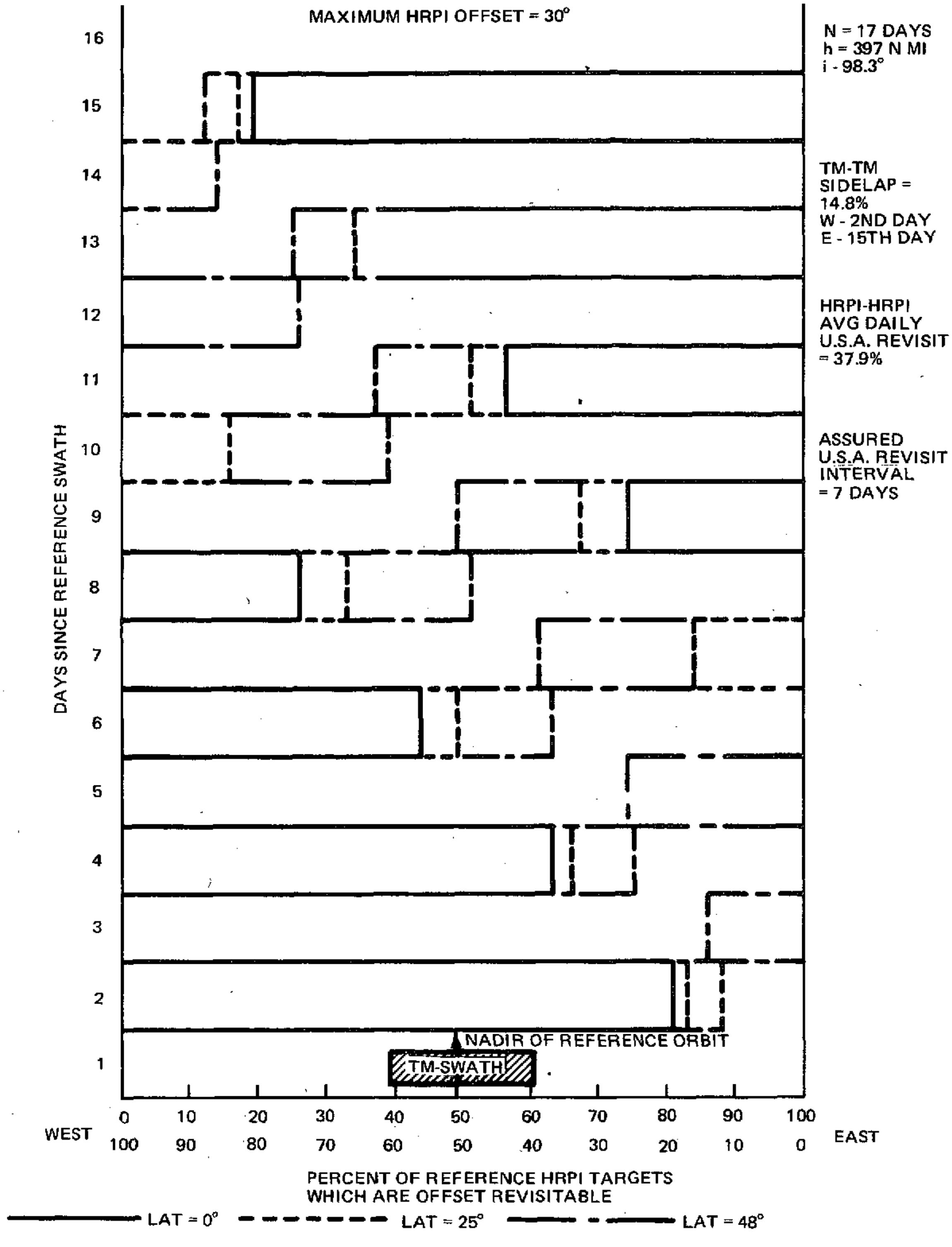


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Fig. 3-7 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 397 N Mi)

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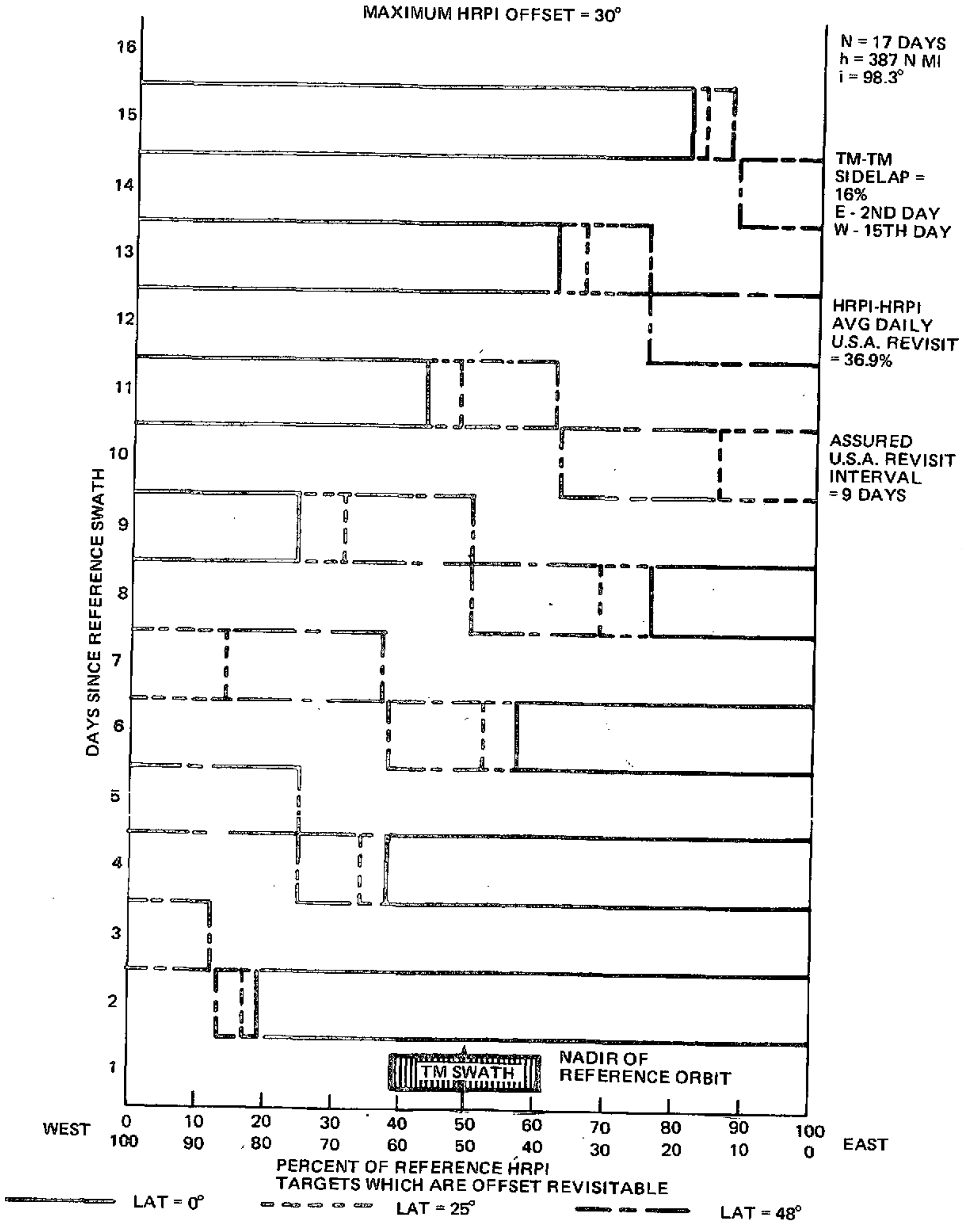


1-8

Fig. 3-8 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 387 N Mi)

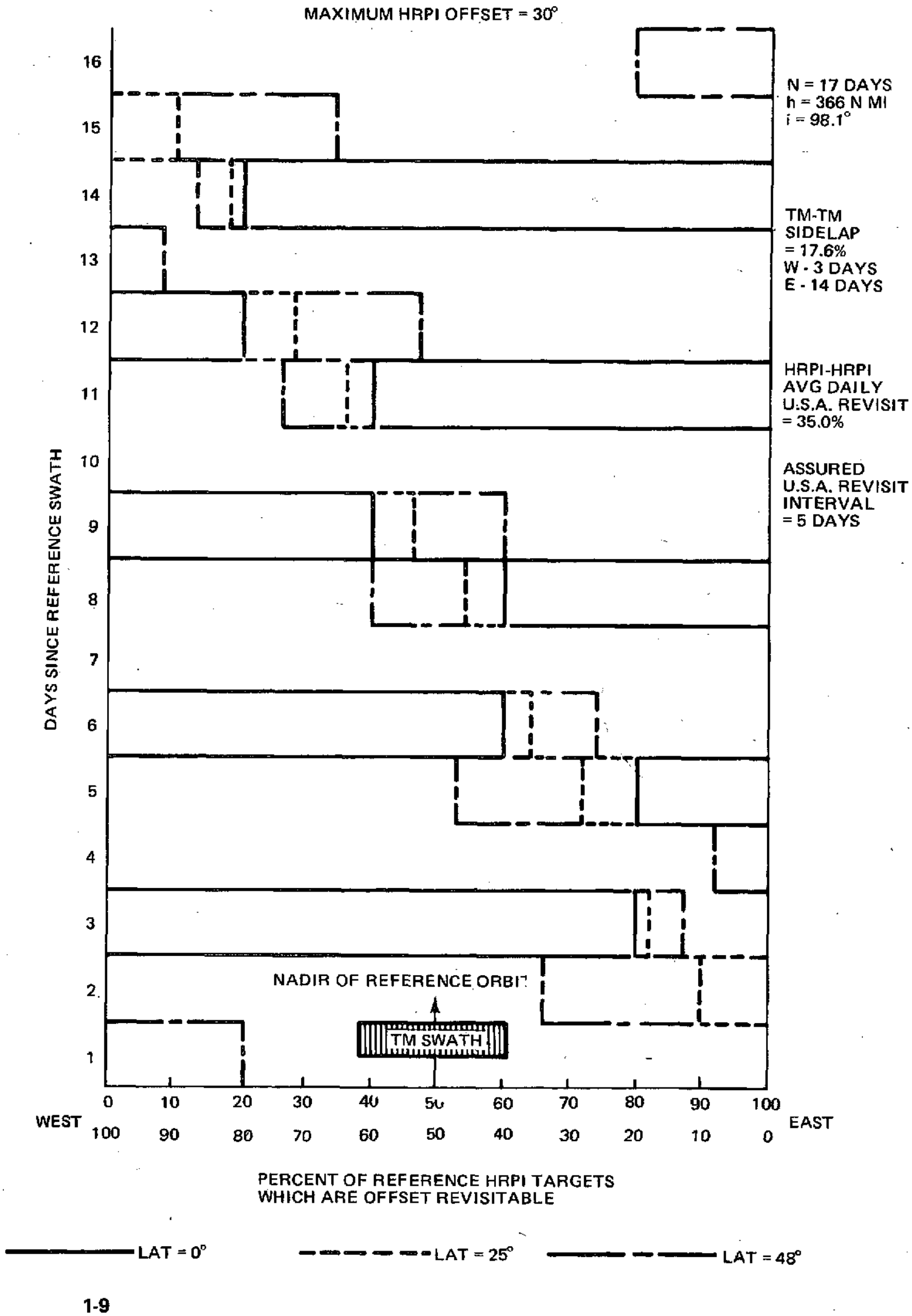
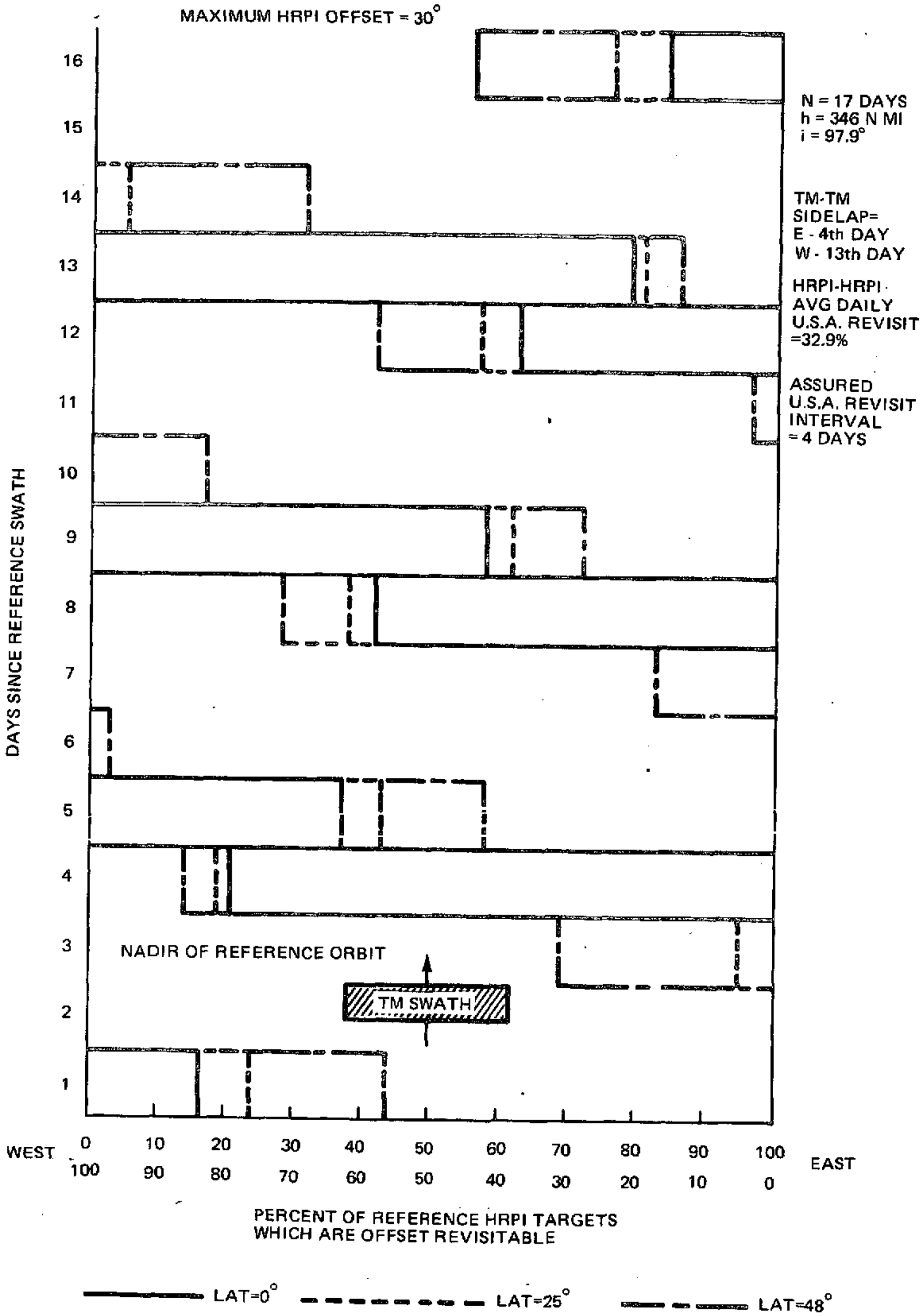
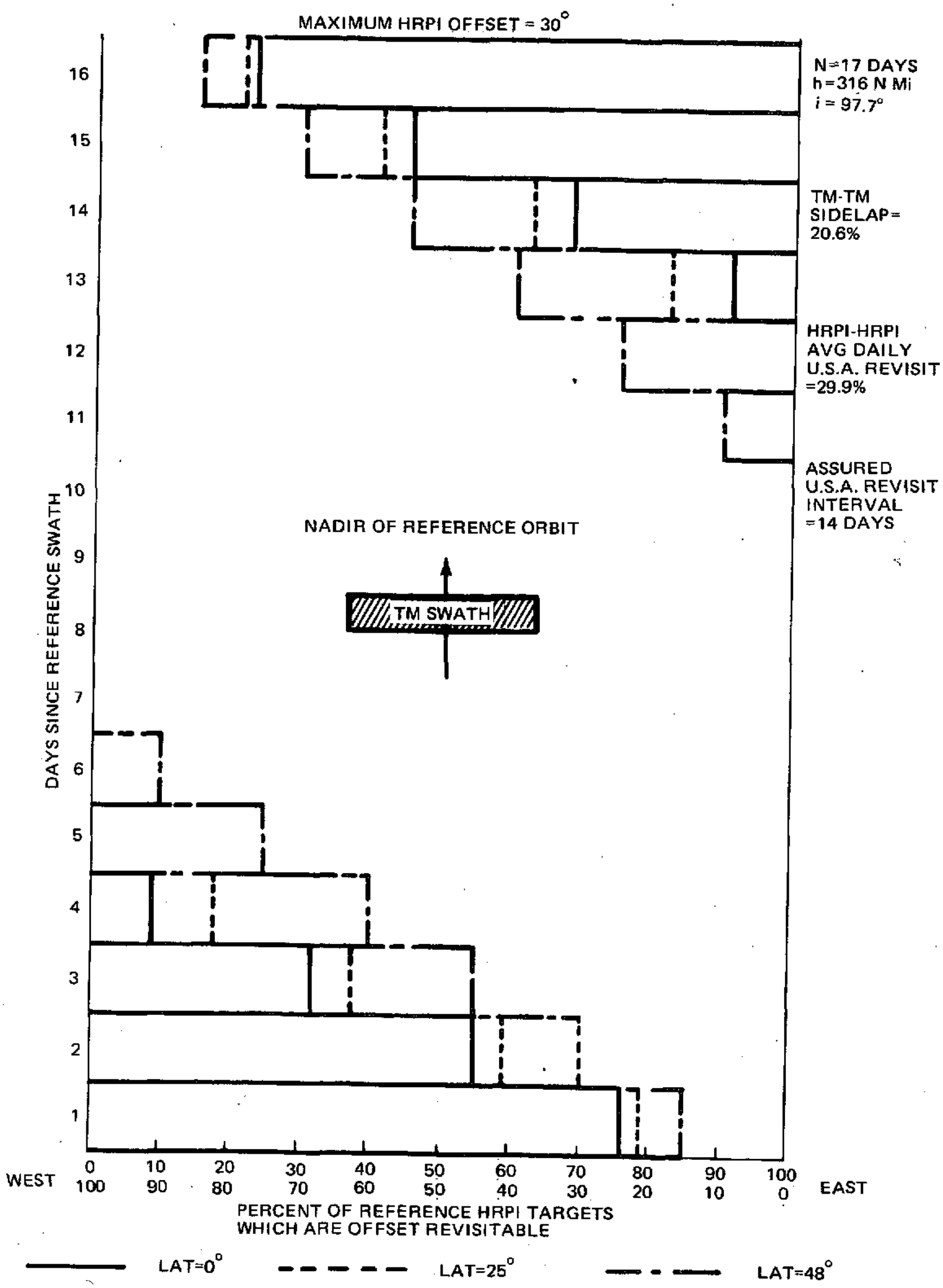


Fig. 3-9 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 366 N Mi)



1-10

Fig. 3-10 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h ≈ 346 N Mi)



1-11

Fig. 3-11 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 316 N Mi)

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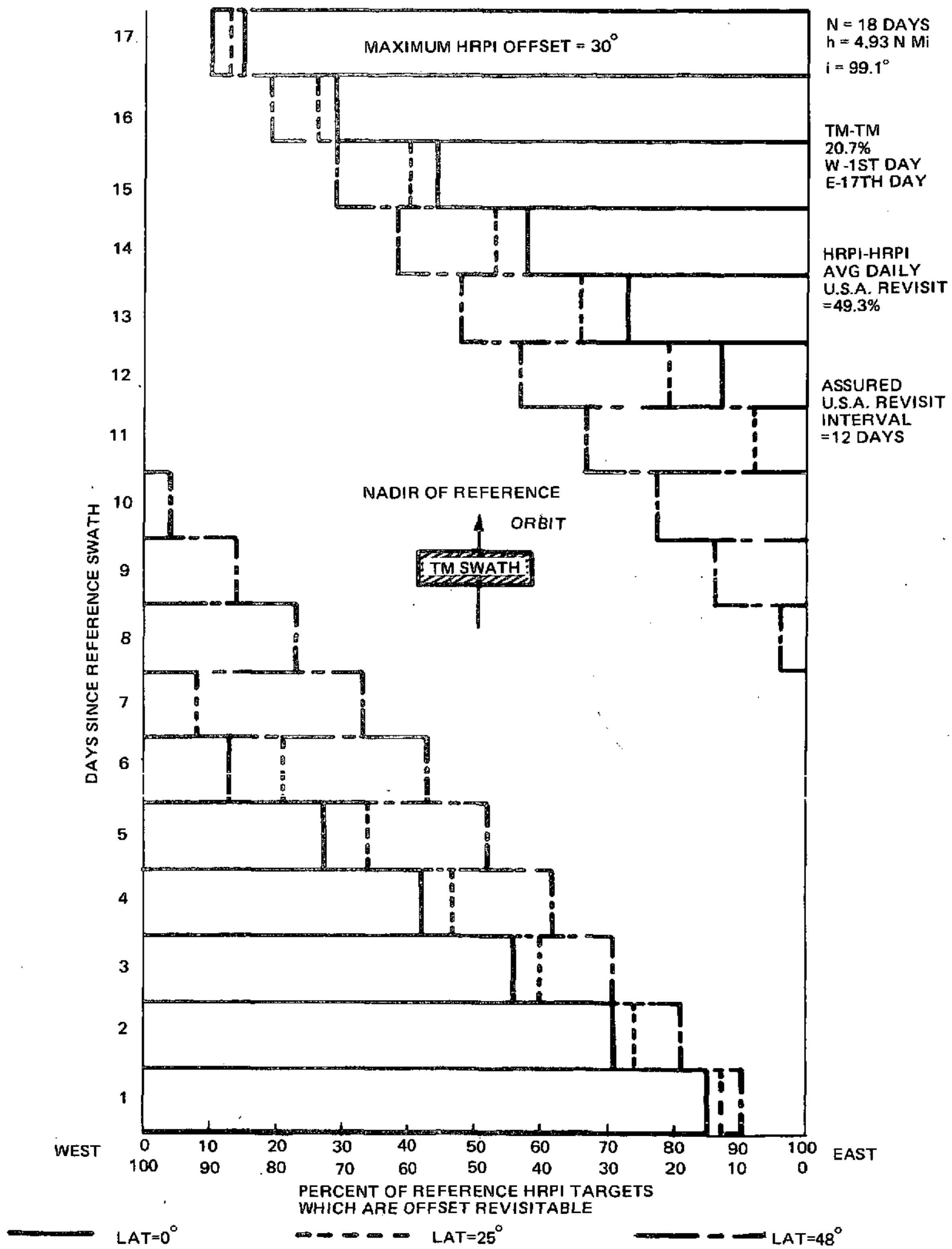
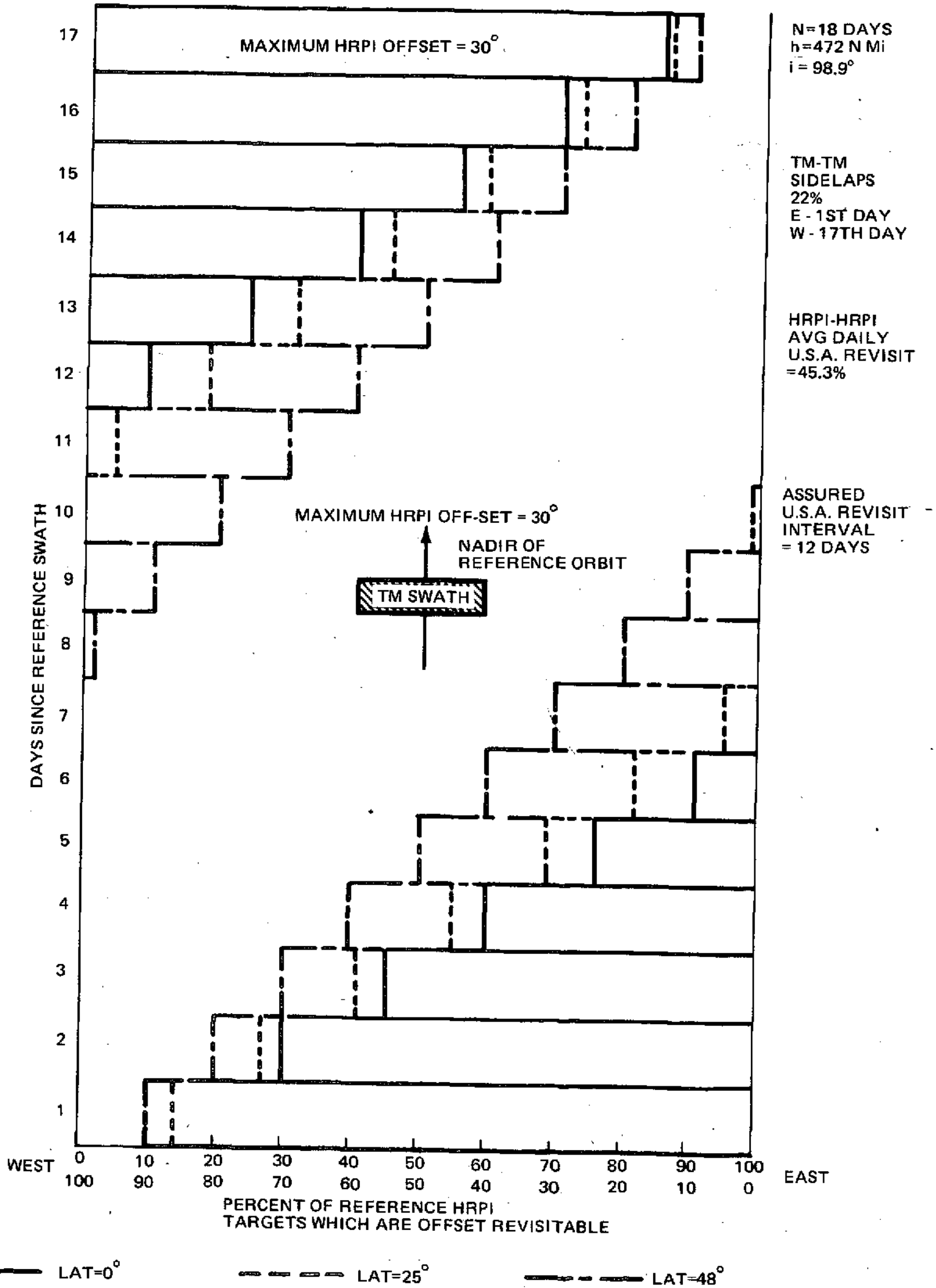


1-12 **Fig. 3-12 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 493 N Mi)**



1-13

Fig. 3-13 High Resolution Pointing Imager and Thematic Mapper Coverage Patterns at Selected Orbit Altitudes (h = 472 N Mi)

Table 3-3 summarizes the results of Fig. 3-5 through 3-9. Using the lower latitude U.S. boundary of 25% as the criterion (overlapping swath coverage increases with increase in latitude), it is seen that of the lower altitude orbits, 345.8, 366.1 and 418.0 n mi provide the most favorable HRPI revisit interval. Altitude 439.2 n mi is equally good in this respect but, as will be shown in Section 6, it is beyond the capability of Shuttle application. Even 418 n mi is marginal.

Selecting these three orbit altitudes (345.8, 366.1, 418.0 n mi), Fig. 3-14 presents plots of cumulative percent of HRPI accessible targets which are revisitable versus target latitude. The effect of opening the HRPI offset angle to 40 deg is also shown in Fig. 3-14. Observe that, whereas an assured 3-day HRPI revisit within all of CONUS is obtained only at altitude 418 n mi when the offset angle is 30 deg., the 366 n mi orbit altitude allows 3-day revisit for latitudes over 40°N, and this improves to 3-day revisit for all latitudes when the HRPI offset is reopened to 40 deg. Altitude 345 n mi does not ensure 3-day revisit within CONUS even with a 40 deg HRPI.

Though altitude 418 n mi is marginally better than 366 n mi for HRPI coverage, the gain in Shuttle utilization at the lower altitude makes the latter an important contender.

TARGET LIGHTING - Within the limits of 9:30 a.m. to 12 noon the selection of DNTD is an EOS user option dictated by experimental needs. As such, the data in this subsection does not bear on the selection of orbit altitude; rather, it serves to provide information on the target lighting at various latitudes, throughout the year, as DNTD is varied.

Table 3-3 Possible Solutions for Swath Overlap Relative to Repeat Cycle Times

MAXIMUM HRPI OFFSET ANGLE 30°

ALT KM	N MI	N	DAYS TO ASSURE HRPI REVISIT LAT = 0°	LAT = 25°	LAT = 48°
813.4	439.2	17	4	4	4 (3)
774.1	418.0	17	5 (3)*	3	3
735.2	397.0	17	9	9 (7)	3
716.0	386.6	17	9	9 (7)	5 (3)
678.0	366.1	17	5	5 (3)	3
640.5	345.8	17	5 (4)	4	4
585.1	316.9	17	14	14 (13)	12 (11)
913.1	493.0	18	13 (12)	12 (11)	9 (8)
874.6	472.2	18	13 (12)	12 (11)	9 (8)

*Number in parenthesis shows days to assure HRPI revisit to at least 90% of HRPI accessible targets if that occurs earlier than the assured (100%) date.

T1-3

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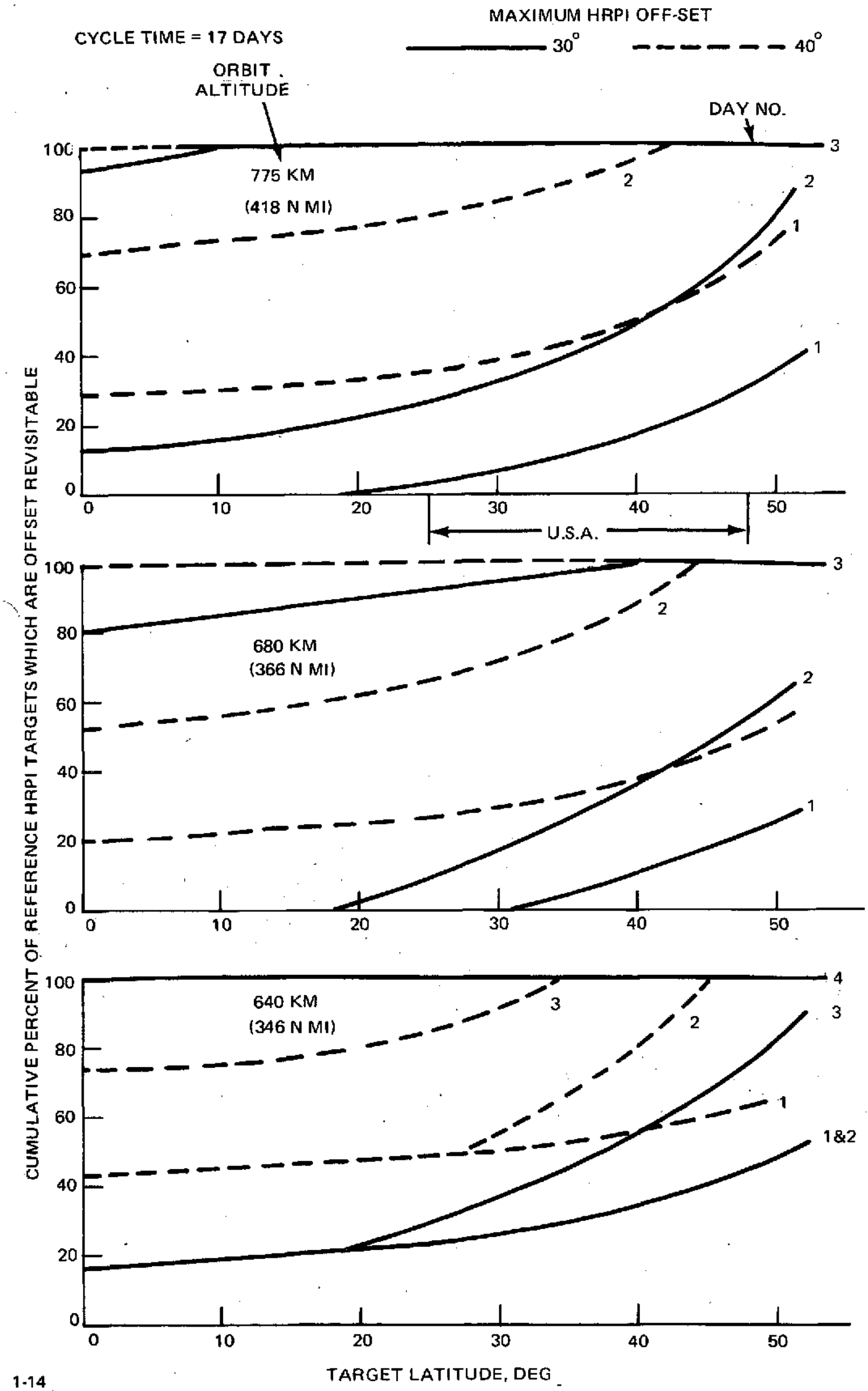


Fig. 3-14 Percent of High Resolution Pointing Imager Targets that are Offset Revisitable

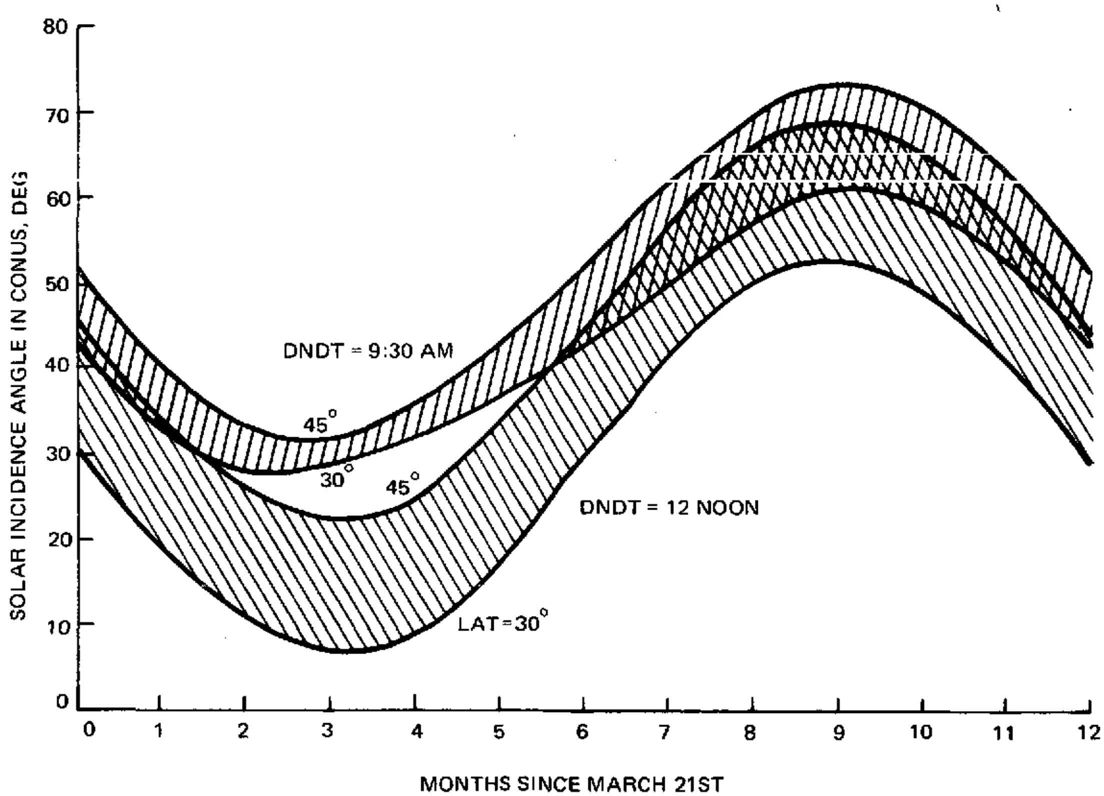
Figure 3-15 shows the solar incidence angle (measured from the vertical at the sub-orbital point) at latitude 30°N and 45°N (i.e., within CONUS) during the course of one year. The two bands on the figure are for the extreme DNTD's of 9:30 a.m. and 12 noon.

Observe that during the year the 9:30 a.m. mission gives rise to a range of incidence angles from about 5 to 70 deg within CONUS.

3.3 ORBIT PERTURBATIONS

The EOS is delivered to mission altitude and the initial adjustments are made to achieve the desired orbit, which is nominally circular and sun synchronous. During the mission, various forces will act to perturb the spacecraft from its initial orbit. The sources of these perturbing forces are:

- Aerodynamic drag
- Spacecraft jet firings
- High-order zonal and tesseral harmonics of the gravity potential function
- Solar radiation pressure
- Solar and lunar gravitation.



1-15

Fig. 3-15 Yearly Solar Incidence Angle at Latitude 30° N and 45° N

3.3.1 AERODYNAMIC DRAG

Aerodynamic drag will most likely be the major contributor to orbit altitude decay. Altitude decay shortens the orbit period and disturbs sun synchronism as described in Subsection 3-1. The change in DNTD by ±1.3 min in three months is insignificant in its effect on lighting conditions, and a greater change over longer periods could be tolerated if lighting alone were the criterion. However, in addition to this small shift in lighting conditions, the attendant shift in imagery becomes excessive and an orbit adjust is required to raise the orbit altitude back to a value which, during the course of decay, will average out to the desired nominal value.

Computation of orbit decay requires some reasonable estimate of the spacecraft ballistic coefficient, which is defined as C_D A/2M. The term C_D is the drag coefficient; A, the effective area which experiences the drag force; and M, the spacecraft mass.

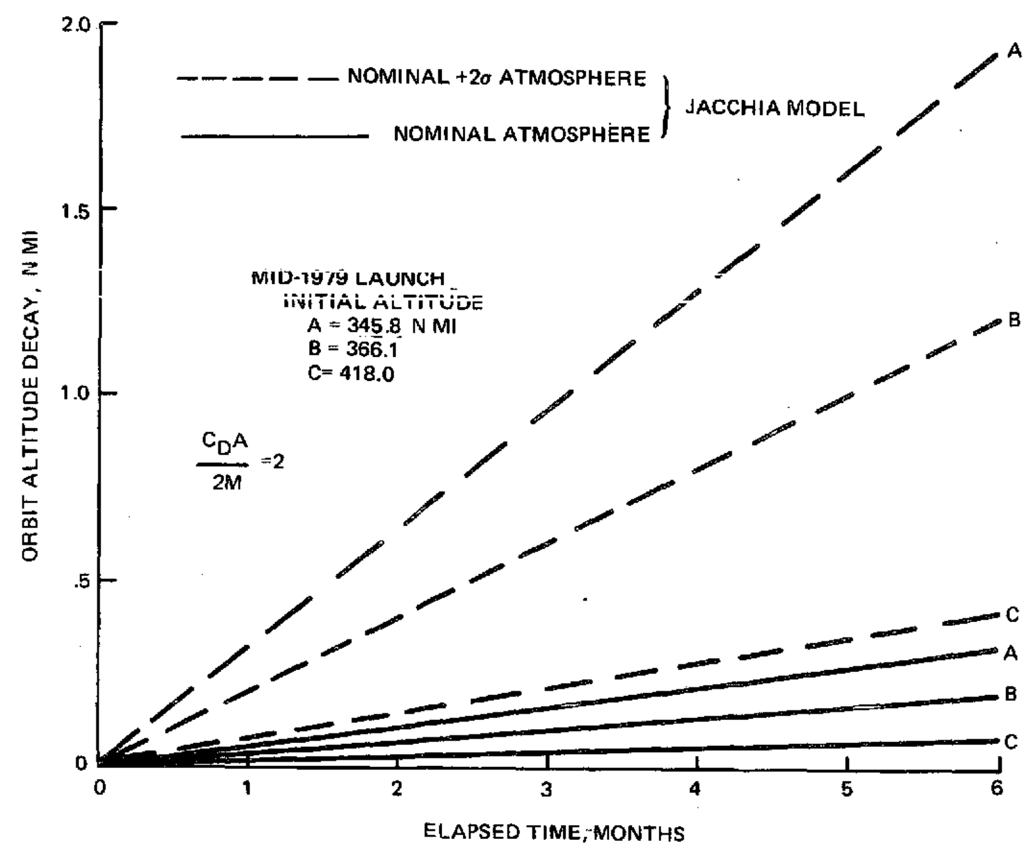
Estimates of spacecraft weight, size, and power requirements given in Report No. 3 indicate that A/M for most configurations will be about 2 ft^2/slug. Past experience with zero drag on vehicles suggests that the drag coefficient, C_D, is likely to fall in the range 1 to 3, with 2 being a good estimate. As a model of the atmosphere, the Jacchia atmospheric model was employed for both a nominal and a nominal $+2\sigma$ air density. The computations were made for an assumed orbit launch in mid-1979. Figure 3-16 shows the altitude decay over a period of six months for orbits with initial altitudes at 345.8, 366.1 and 418.0 n mi, assuming $C_D = 2$. The difference in the effect of the nominal and the nominal $+2\sigma$ atmosphere is seen to be significant. However, though prudence may dictate the use of the latter nominal $+2\sigma$ atmosphere, in determining the orbit adjust fuel requirements, it would appear to be unnecessarily pessimistic in estimating the expected orbit adjust frequency.

The principal effect of the orbit decay (and period reduction) is to separate the corresponding tracks of two successive repeat cycles; tracks which, without this orbit decay would be exactly overlaid, one upon the other. This result, a nodal sideslip, is presented in Fig. 3-17 for a 366-n mi orbit with spacecraft ballistic coefficient 2. From Fig. 3-16 and 3-17 we conclude that for a nominal atmosphere, after three months of operation the atmospheric drag will reduce the altitude by 0.10 n mi thereby inducing a nodal sideslip of 40 n mi from its starting position. (This corresponds to ± 20 n mi on Fig. 3-17 since the variation is considered to be about a nominal node selected at the midpoint of the range.) With a repeat cycle time of 17 days, this works out to about an 8-n mi shift between corresponding tracks over the time interval of one repeat cycle.

If the nominal + 2σ atmospheric model is adopted for computation, Fig. 3-17 shows that a constraint of ±20 n mi in nodal sideslip requires an orbit adjust approximately every five weeks, which is not considered to be particularly excessive.

The effect on altitude decay of varying the ballistic coefficient can be seen from Fig. 3-18.

Adjustment of the orbit is, in principle, achieved with two jet firings on a Hohmann transfer orbit. The total ΔV required depends upon the altitude loss at the time the correction is made. Assuming, again, a 366-n mi orbit with spacecraft ballistic coefficient

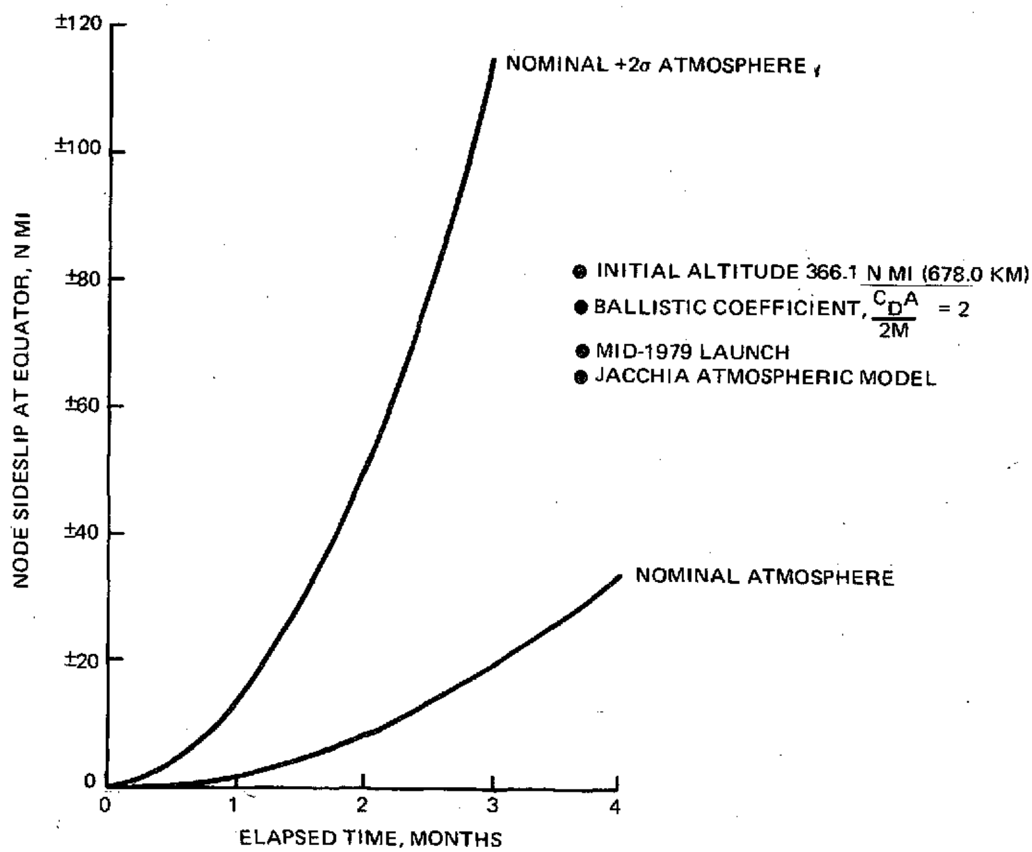


1-16 Fig. 3-16 Six-Month Altitude Decay for Three Selected Orbit Altitudes

2, the nominal atmosphere computation shown on Fig. 3-19 indicates a three-month ΔV correction requirement of about 0.3 fps for a total of 2.1 fps over a two-year spacecraft lifetime (seven adjusts). The nominal $+2\sigma$ atmosphere requires a ΔV of 0.8 fps every 1.25 months for a total two-year ΔV of 15.2 fps (19 adjusts). In neither case does this represent a particular problem in providing the necessary orbit adjustments.

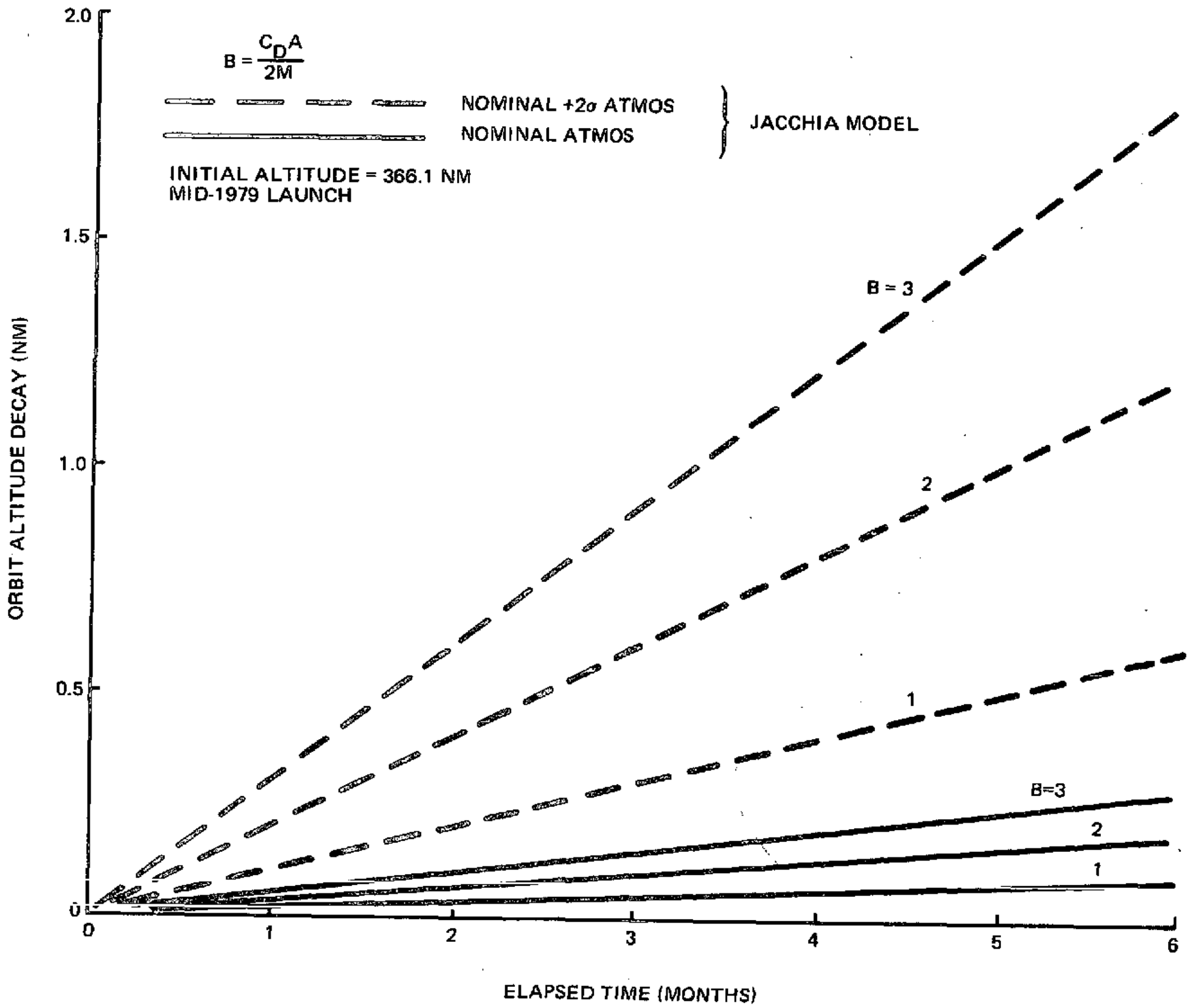
3.3.2 OTHER PERTURBATIONS

Analyses of the ERTS orbit has shown a significant component of altitude loss which is presently attributed to operation of the attitude control system. On the ERTS, uncoupled thrusters, which provide momentum wheel unloading, also supply a net force in the direction opposed to the spacecraft velocity vector, thereby achieving the effect of increased aerodynamic drag. The magnitude of this contribution correlates well with the utilization of pitch gate jet operation which, in turn, seems to depend upon the extent of magnetic tape recording required during the orbit. On EOS, this source of disturbance is practically



1-17

Fig. 3-17 Node Sideslip Due to Aerodynamic Drag

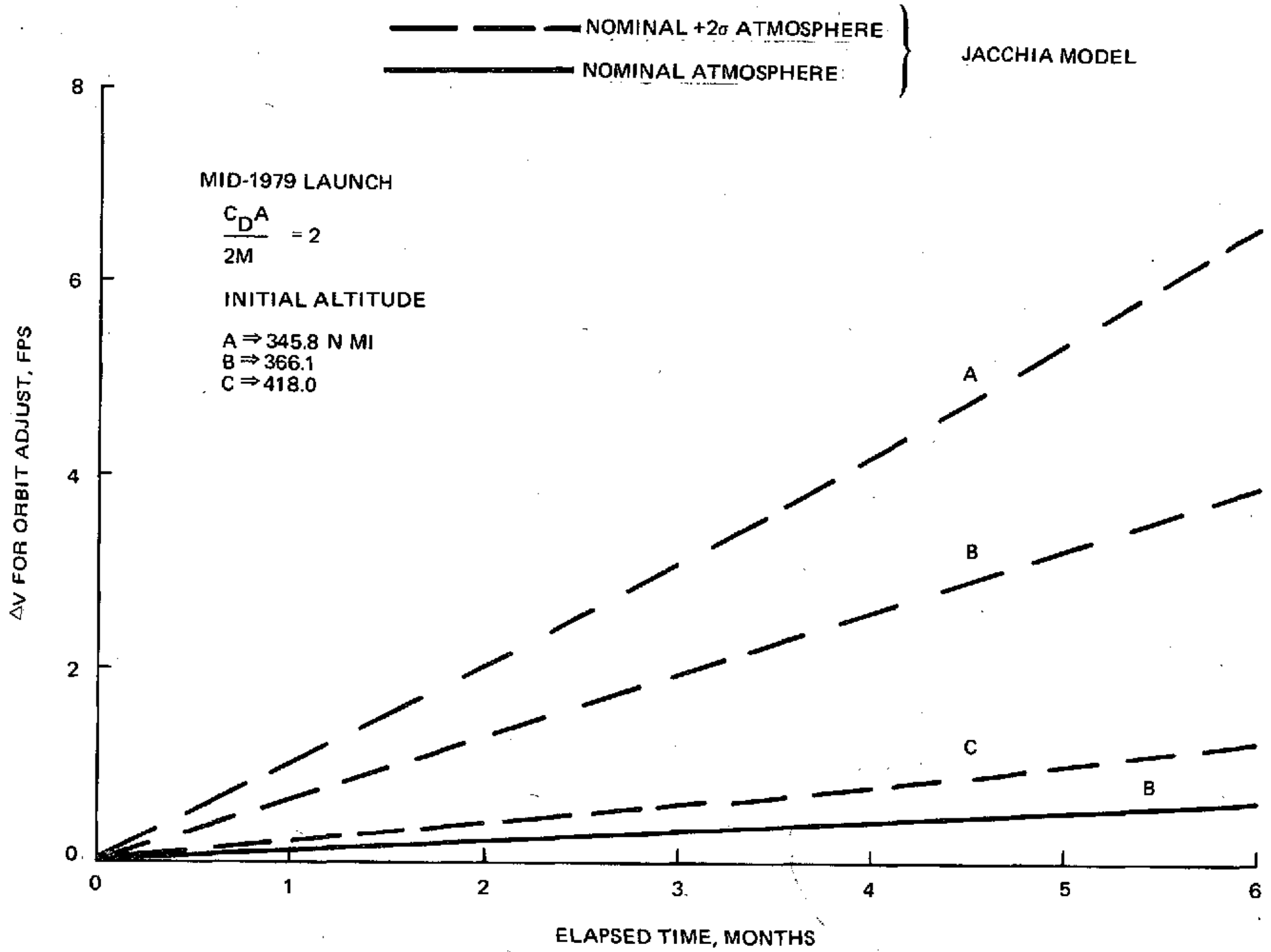


1-18

Fig. 3-18 Delta Ballistic Coefficient Effect on Orbit Altitude Decay

eliminated in two ways. First, magnetic unloading of the momentum wheels is the principal device employed. Creation of a magnetic dipole avoids the imposition of a net translatory force. Second, if momentum storage should become excessive and magnetic unloading is insufficient, a coupled jet system is used and, again, net forces are effectively eliminated.

Solar radiation pressure effects, like aerodynamic drag, are a function of the ratio, A/M, where A is the effective spacecraft area which "intercepts" the solar radiation and M is the spacecraft mass. The force on the spacecraft due to solar radiation acts to induce a change in orbit eccentricity and leaves the semi-major axis (and therefore orbit period)



1-19

Fig. 3-19 Typical Delta Velocity for Orbit Adjustments to Achieve Design Spacecraft Lifetime

essentially unchanged. Assuming a solar radiation pressure of 2×10^{-7} lb/ft^2 and A/M ~ 2 ft^2/slug, an initially circular orbit at 366-n mi altitude should, after three months, exhibit eccentricity of the order 0.75×10^{-6}. At this eccentricity the altitude over the course of one period will deviate at most ± 0.03 n mi (approximately 55 meters) from its initial value. This falls within the perturbation attributed to aerodynamic drag and would be corrected as part of the overall orbit adjustment. Other perturbations, resulting from gravitational anomalies, are of much lower order and will be nulled as each orbit adjustment is executed without any ΔV requirement beyond that estimated for a nominal $+2\sigma$ atmosphere.

S/C IMPACT ON
ORBIT SELECTION

4 - SPACECRAFT IMPACT ON ORBIT SELECTION

The choice of orbit bears upon the spacecraft design in so far as the altitude and DNTD impacts the thermal and lighting environment at the spacecraft location. This section provides data on solar lighting on the vehicle and solar panels. Impact on pertinent subsystem designs is summarized; a fuller discussion is given in Report No. 3.

4.1 SOLAR LIGHTING

The spacecraft passes into earth's shadow once during each orbit. The time spent in shadow varies with such factors as orbit altitude, inclination, descending node time-of-day (DNTD), and day of the year. Increasing altitude increases the time spent in shadow. For a given altitude, a DNTD of 12:00 noon provides the longest shadow time per orbit.

Table 4-1 shows the maximum and minimum in-shadow durations for a range of DNTD from 9:30 a.m. to 12:00 noon when the orbit altitude is held at 366 n mi. The maximum time in darkness is seen to be 35.33 min, about 36% of the orbit period (98.33 min).

The solar panels, being torqued about an axis perpendicular to the orbit plane, maintain an almost constant orientation relative to the sun. Though the motion of the sun and the orbit node have nominally the same value (giving rise to a sun-synchronous orbit), they move in different planes: the sun along the ecliptic and the orbit node along the equator. This difference in motion gives rise to a yearly variation in the angle of incidence of the sun's rays on the solar panel. The variation is, therefore, a consequence of the sun's north-south motion with the changing seasons.

With the mission DNTD selected it is possible to determine the panel orientation which minimizes the solar incidence angle on the panel. The incidence angle is defined as the angle between the solar rays and the panel normal. The angle that the panel normal should make with the orbit plane, referred to as the panel knuckle angle, is shown in Table 4-2 for various DNTD. With these knuckle angle settings, the solar ray's maximum deviation from normal panel incidence will be minimized. These deviations are of the order of 5 deg and result in no significant degradation of solar power capture.

The spacecraft design requires the solar panel to be on the same side of the orbit plane as is the sun. In the proposed design, this is the case when the DNTD is earlier than 12:00 noon. When the DNTD is selected to be in the afternoon, the vehicle, after orbit insertion, is rotated 180 deg about its vertical axis to keep the panel on the sun side. The consequent "backward" motion of the spacecraft presents no problem to any of the spacecraft functions while collecting and telemetering data. To perform an orbit adjust, however, the spacecraft would either have to be returned to its normal orientation or pitched 180 deg to an upside-down orientation before activating the orbit adjust thrusters.

The impact on the spacecraft design of having the sun irradiate the sides not designed for that environment will be investigated in later studies. A solution which requires no "backward" flight for afternoon DNTD is to perform a nighttime launch at WTR and have the spacecraft make its daylight passes over CONUS moving northward. This, however, impacts the gathering of imagery and would require user approval.

4.2 IMPACT ON SPACECRAFT SUBSYSTEM DESIGN

Solar irradiation of the spacecraft has both desirable and adverse effects. Whereas the solar panel requires solar energy to perform its function, other subsystems are more satisfactorily operated away from direct solar irradiation. This subsection briefly summarizes results of studies on thermal and power supply effects due to varying DNTD. A detailed discussion is given in Report No. 3, Appendix D.

Table 4-1 Yearly Range of Occultation Durations

DNTD	MINIMUM	MAXIMUM
9:30 AM	30.45 Min	33.15 Min
10:00	32.42	34.08
10:30	33.62	34.73
11:00	34.47	35.14
11:30	35.00	35.32
12:00 noon	35.27	35.33

T1-4

Table 4-2 Panel Knuckle Angle and Yearly Maximum Incidence at Panel

DNTD	KNUCKLE ANGLE	MAXIMUM INCIDENCE
9:30 AM	34.8°	5.75°
10:00	28.0	5.35
10:30	21.0	5.15
11:00	14.0	4.95
11:30	7.0	4.85
12:00 noon	0.	4.80

T1-5

4.2.1 ORBITAL HEAT FLUX STUDY

The successful thermal design of an EOS requires precise control of the operating temperature of the instruments and subsystem equipment. Spacecraft temperatures depend on the heat balance between vehicle external surfaces and its environment, which include the following external sources:

- Radiation emitted by the sun
- Radiation emitted by the earth
- Solar radiation reflected by the earth.

Knowledge of the extremes of environment heat fluxes for the range of EOS orbits is, therefore, of primary importance in the design of the temperature control system.

4.2.1.1 FLUX MODEL AND CASES CONSIDERED

Transient and orbital average heat fluxes were generated for the range of sun-synchronous orbit parameters covering the EOS mission. External heat fluxes, consisting of direct solar, earth albedo and earth IR, were determined using Grumman's orbital heat flux program. A total of 14 computer runs were made for the following thermal environment constants and combinations of parameters:

- Thermal Environment Constants:
  - Solar Constant: Vernal Equinox - 430 BTU/hr ft^2; Winter Solstice - 444 BTU/hr ft^2; Summer Solstice - 415 BTU/hr ft^2
  - Albedo Constant: .30
  - Earth Emission: 75 BTU/hr ft^2
- Orbit Parameters:
  - Orbit Altitudes: 300, 366, 400, 500 n mi (circular)
  - Orbit Inclinations: 97.55, 98.09, 98.30, 99.10 deg (South-heading)
  - Descending Node Times of Day: 0930, 1030, 1200, 1330 hr
  - Times of Year: Vernal Equinox, Winter Solstice, Summer Solstice.

To provide flux data for the EOS/Titan and for the two EOS/Delta spacecraft configurations considered, a generalized 30-surface flux model, shown in Fig. 4-1, was developed.

Table 4-3 summarizes the absorbed external heat fluxes for each subsystem module location for all 14 computer runs. The orbit condition at which maximum and minimum fluxes occur is also identified.

4.2.1.2 EFFECT OF ORBIT PARAMETERS ON THERMAL HEAT FLUXES

The impact of sun-synchronous orbit parameter variations on the thermal control system can best be evaluated by examination of the absorbed environmental heat fluxes for the various subsystem modules external skins. A typical EOS/Delta configuration, with Alzak skins on the EPS, ACS and C&DH modules, is examined for this purpose: however, the results are applicable for all spacecraft configurations.

Figure 4-2A shows the variation in absorbed heat flux as a function of orbit altitude with DNTD of 0930 and 1200 as the parameter. Examination of this curve shows a very slight reduction in absorbed heat flux as altitude increases from 300 to 500 n mi. Figure 4-2B shows the variation in absorbed heat flux as a function of DNTD for a 366-n mi altitude. These results show a small variation in absorbed heat flux for the earth facing C&DH module, and significantly larger variations for the EPS and ACS modules due primarily to direct solar flux inputs. Design of the EOS for a DNTD varying between 0930 and 1330 has an extreme impact on instrument passive cooler feasibility and a significant impact on subsystem module thermal control. The passive cooler must be located on a surface receiving essentially zero heat flux; no single surface location on the spacecraft had this property for both morning and afternoon orbits. Thus, consideration of passive cooler flux requirements dictate either a morning (i.e., 9:30 to 11:30) DNTD or an afternoon (12:00 to 2:30); however, operating a single instrument in both morning and afternoon DNTD orbits is not possible.

The non-earth viewing subsystem modules are also affected by the range of DNTD. Although an acceptable thermal design of these modules is possible for both morning and afternoon orbits, penalties for this capability will be incurred in terms of heater power (array cost and weight) and the cost and weight of active thermal control to reduce the heater power.

Thus, by limiting DNTD to a morning-only orbit, the possibility of using passive coolers for the instruments requiring them is established and the simplification of thermal control hardware is achieved on the non-earth viewing subsystem modules. From Fig. 4-2B, it is obvious that any further reduction in the design range of DNTD results in an improved thermal design environment.

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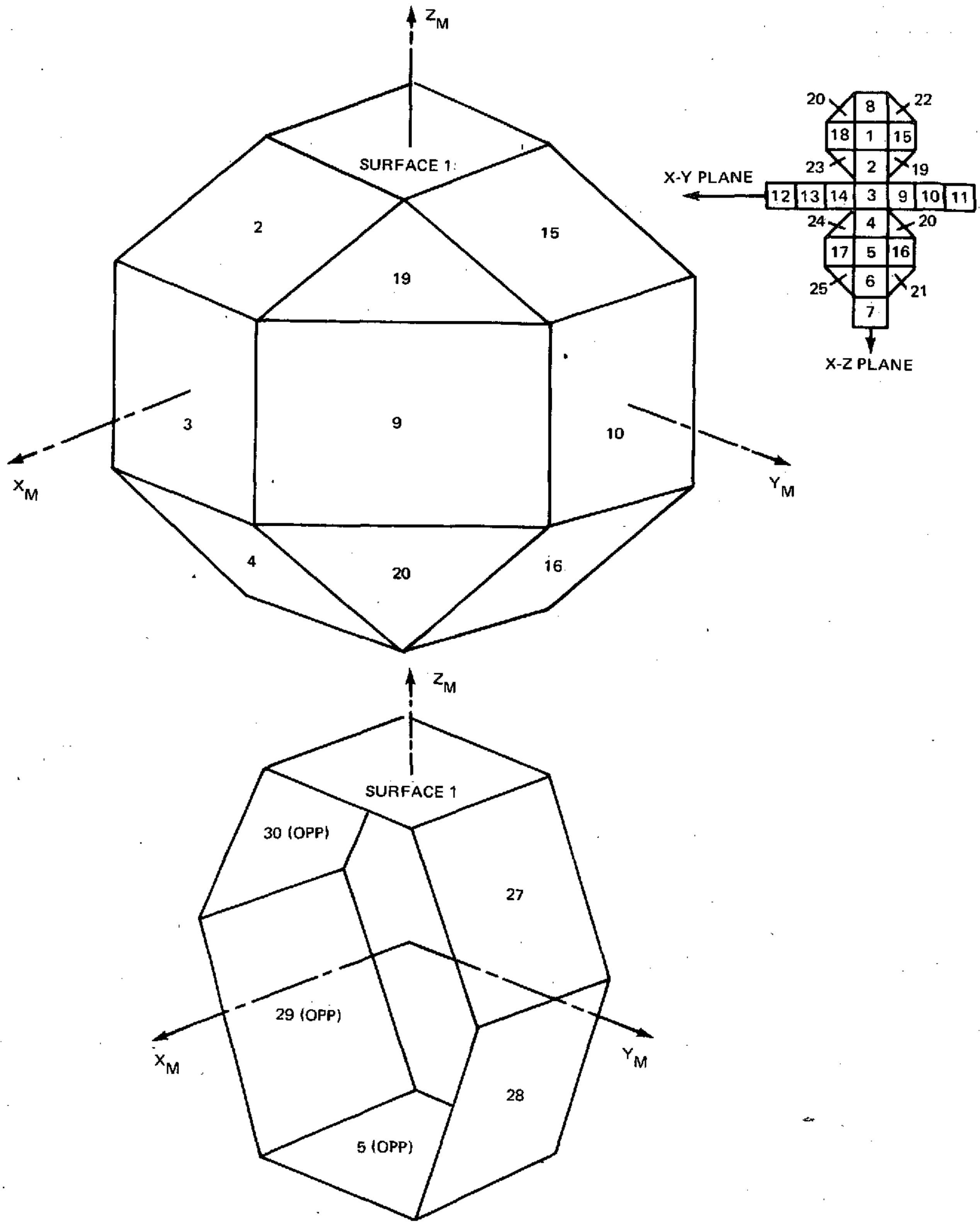

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Fig. 4-1 Orbital Heat Flux Model Surface Numbers

Table 4-3 Summary of Orbital Heat Fluxes Absorbed by Modules External Absorbed Heat Flux[1] (BTU/Hr Ft)

FLUX RUN NO.	DESCENDING NODE TIME OF DAY	SEASON	ALT. (NM)	SURF. 1	SURF. 5	SURF. 10	SURF. 13[2]	SURF. 27	SURF. 28	SURF. 29[2]	SURF. 30[2]
1	0930	VER. EQ.	366	52.0	16.4	14.2	40.9	28.4	3.9	33.8	44.8
2	1030	VER. EQ.	366	52.2	19.0	14.4	30.6	28.9	5.4	26.2	36.0
3	1200	VER. EQ.	366	52.3	20.4	14.6	14.6	30.2	14.2	14.2	30.2
4[3]	1330	VER. EQ.	366	52.2	19.0	30.6	14.4	36.0	26.2	5.4	28.9
5	0930	VER. EQ.	300	53.2	16.4	15.4	41.5	29.9	4.5	34.1	45.6
6	1200	VER. EQ.	300	53.6	20.4	15.8	15.8	31.5	14.9	14.9	31.5
7	0930	VER. EQ.	400	51.4	16.4	13.6	40.7	27.7	3.6	33.6	44.5
8	1200	VER. EQ.	400	51.6	20.4	14.0	14.0	29.5	13.9	13.9	29.5
9	0930	VER. EQ.	500	49.7	16.5	12.2	39.9	25.8	2.9	33.2	43.6
10	1200	VER. EQ.	500	49.7	20.4	12.6	12.6	27.8	13.2	13.2	27.8
11	0930	WIN. SOLS.	366	52.2	16.9	14.2	42.2	28.5	3.9	35.0	45.8
12	1200	WIN. SOLS.	366	52.7	21.2	14.7	17.1	29.9	13.0	16.3	30.9
13	0930	SUM SOLS	366	51.8	17.2	14.3	35.0	28.6	3.9	29.3	39.2
14	1200	SUM SOLS	366	52.2	19.8	16.8	14.6	30.6	15.4	12.4	29.7

(1) ORBITAL AVERAGE SOLAR, ALBEDO AND EARTH EMISSION ABSORBED BY SURFACE WITH $\alpha_s = 0.15$ AND $\epsilon_{TH} = 0.75$

(2) ADDITIONAL SOLAR ARRAY FLUX REQUIRED FOR THIS SURFACE

(3) 1330 ORBIT HOUR ANGLE NOT CONSIDERED FOR DESIGN

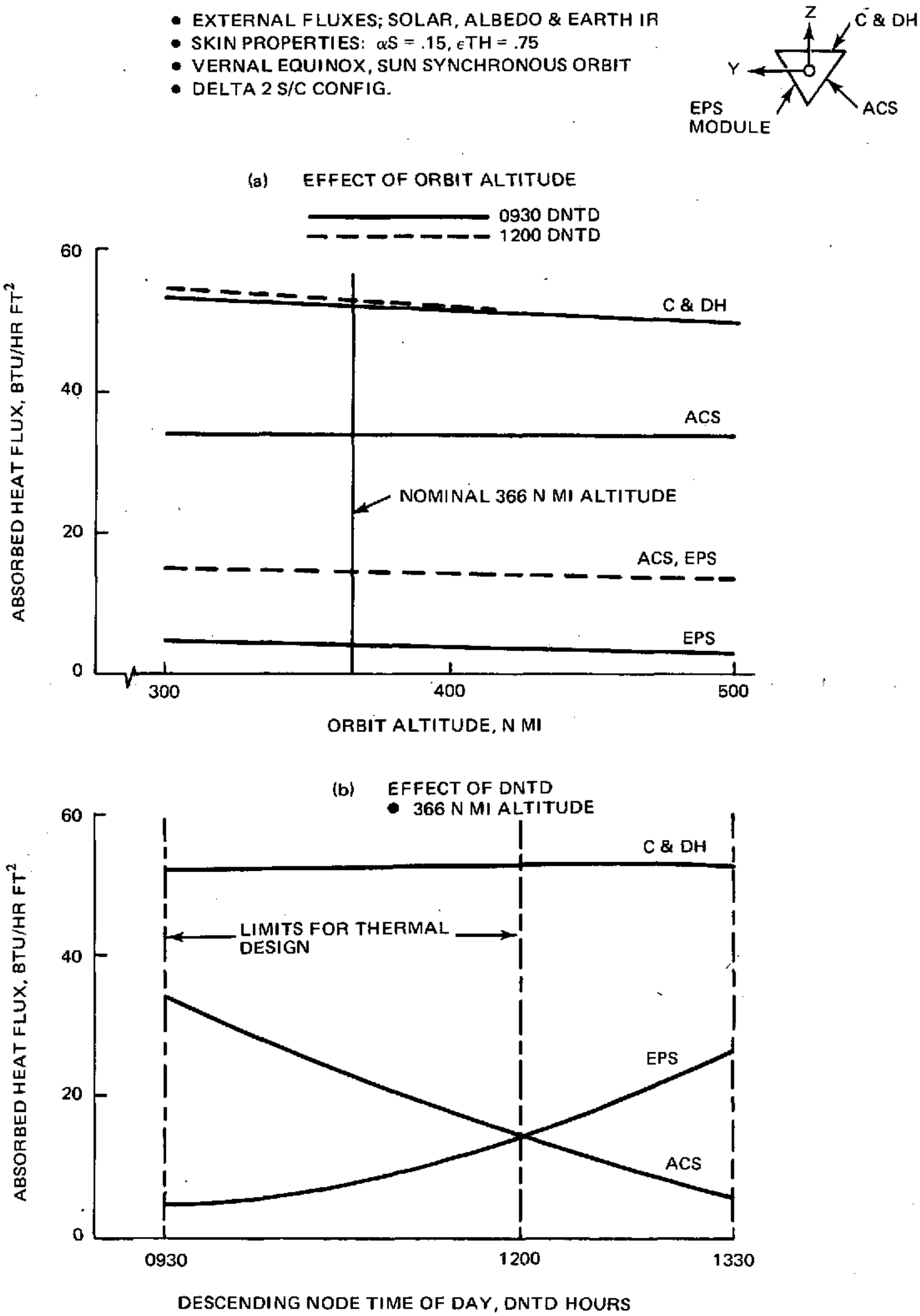
MINIMUM FLUX (oval)

MAXIMUM FLUX (box)

FLUX MODEL SURFACE NO.

MODULE	TITAN	DELTA 1	DELTA 2
EPS	10	27	28
C&DH	13	30	1
ACS	5	5	29

T1-6



1-21 **Fig. 4-2 Effect of Orbit Parameters on Module Absorbed Heat Flux (Delta 2 S/C Configuration)**

4.2.2 ELECTRIC POWER SYSTEM IMPACT

Orbital parameters will in general affect the EPS in the following manner:

- Ratio of Dark-to-Light Duration - This important EPS design relationship has a direct effect on both battery and solar array sizing. For the range or orbits considered for EOS, the variation in the nominal dark to light ratio of 36/65 will have a negligible effect on EPS sizing.
- Solar Array Radiation Environment - In general, the higher the orbit and the inclination, the greater the amount of power degradation in the solar array. However, given a nominal sunsynchronous orbit (inclination), the effect of raising the orbital altitude from 300 to 500 n mi will result in only about 1% additional array power degradation, which is considered negligible.
- Solar Array Orientation - The DNTD will have a direct influence on the solar array orientation requirements. For example, a 12:00 noon orbit would require a rotating array while a terminator orbit (6:00 a.m/p.m) would not. For the relatively large spacecraft power requirements of EOS, and a DNTD later than approximately 8:00 a.m., it would be advantageous to provide array rotation.

An additional consideration involving the DNTD and the EOS earth pointing requirement, is that spacecraft shadows on the array should be avoided. The array should be mounted on the sun-side of the spacecraft to satisfy this requirement. This would tend to rule out afternoon DNTD's unless provisions are made to either roll or yaw the spacecraft 180 deg, or relocate the array and drive to the sunside.

MISSION TRACKING
COVERAGE

5 – MISSION TRACKING COVERAGE

The altitude of the EOS mission impacts the ground station tracking history and the time interval available for data acquisition. In this section consideration is given to the instance where direct data transmission during CONUS and Alaska operations is acquired through (1) two stations only: Sioux Falls, S. D. and Fairbanks, Alsaka and (2) three stations: Sioux Falls being replaced by Goldstone, Calif and NTTF, Md.

Analyses were performed for two candidate EOS orbital conditions: a 366 n mi (678 km) circular orbit altitude inclined at 98.1 deg and a 493 n mi (913 km) circular orbit altitude at 99.1 deg inclination.

Presented in Fig. 5-1 is a map showing the ground track profiles and resulting ground track coverage for the 493-n mi circular orbit mission. The ground track coverage is based on minimum elevation tracking angles as follows:

- Fairbanks, Alaska – 5 deg
- Goldstone, Calif – 5 deg
- NTTF, Md. – 5 deg
- Sious Falls, N. D. – 2 deg

As is readily shown, Complete CONUS coverage is achieved by either the Sioux Falls station alone or by the two stations, Goldstone and NTTF operating jointly. At this mission altitude, the longest continuous data transmission is approximately 14 min, providing data for EOS operations far into Canada and Mexico as well as over CONUS. The Fairbanks Alaska station provides complete coverage for EOS operations over Alaska and the northwestern part of Canada.

The map in Fig. 5-2 shows similar contours for the aforementioned stations when the mission altitude is 366 n mi. Observe that the combined Goldstone and NTTF stations provide complete data acquisition for all of the CONUS operations, while the Sioux Falls station alone still provides essentially complete CONUS coverage.

The southern extreme of Florida, in particular, the Keys, may not be included within Sioux Falls coverage for TM imagery, but this region is easily scanned with a HRPI instrument during a Florida overpass. The maximum ground track interval at altitude 366 n mi is 11.4 min.

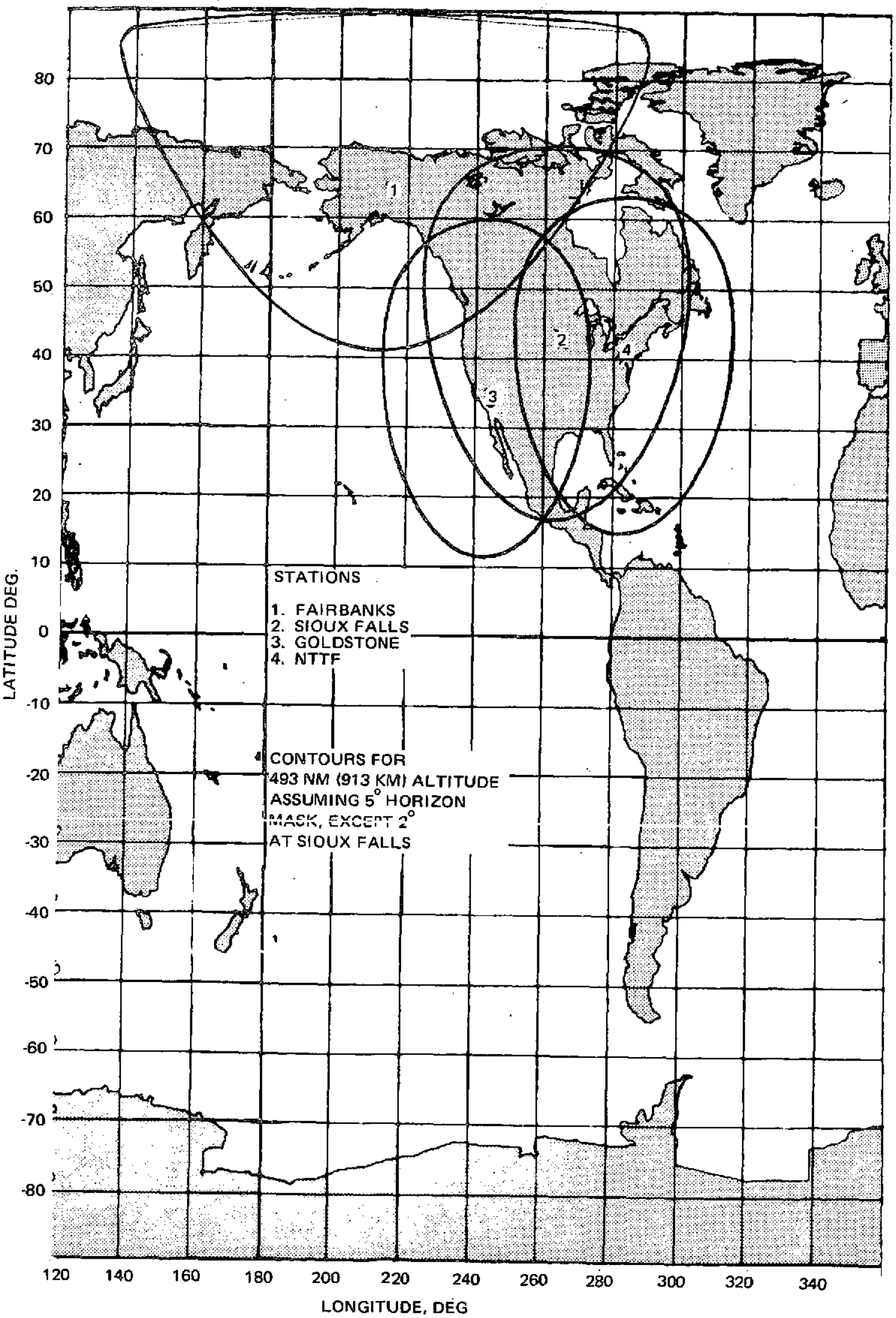


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Fig. 5-1 Ground Track Coverage — 493-Nautical Mile Circular Orbit

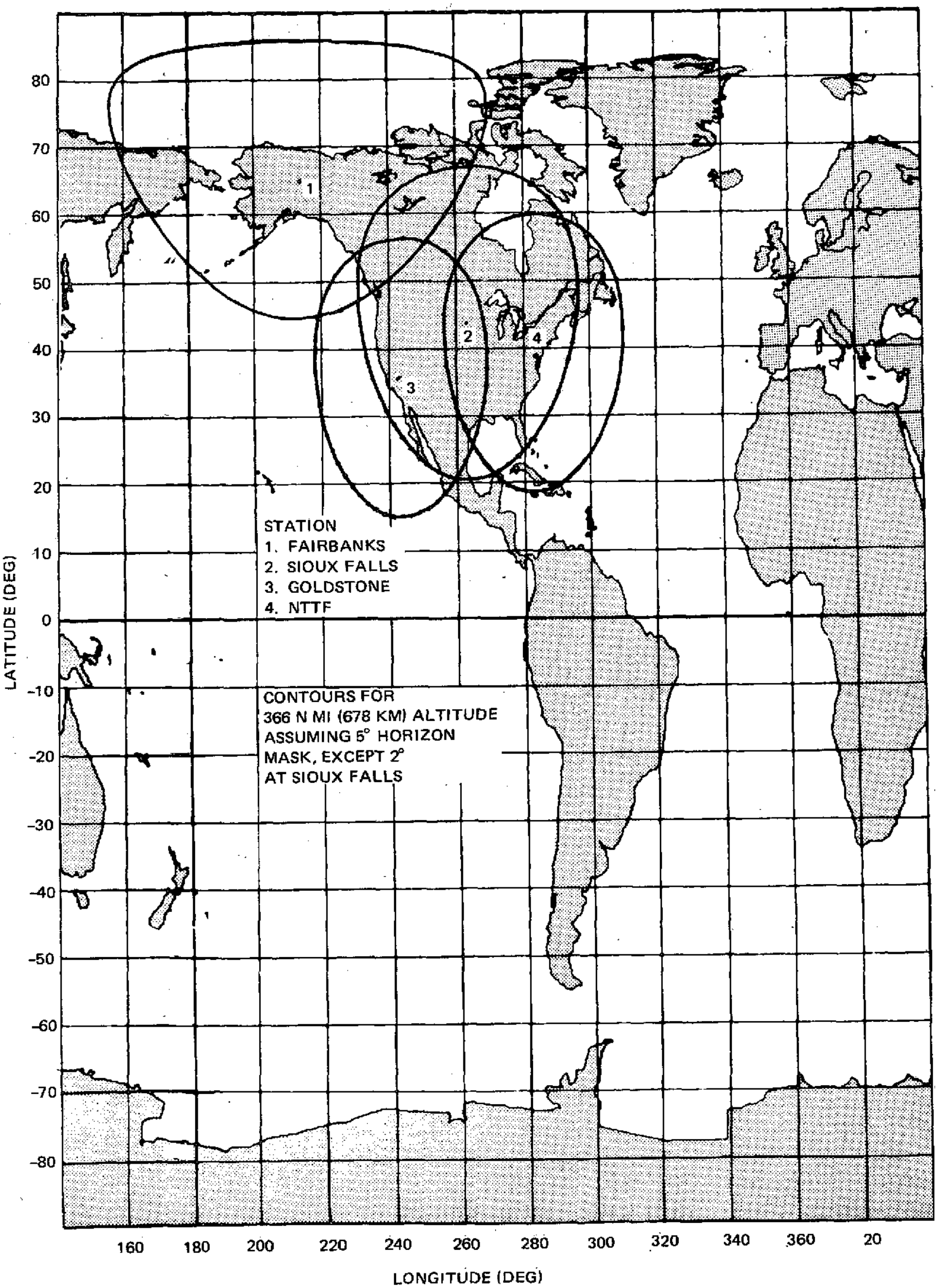

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Fig. 5-2 Ground Track Coverage – 366-Nautical Mile Circular Orbit

The conclusion reached in these studies is that the Fairbanks station, together with either the Sioux Falls station or the combined Goldstone and NTTF stations, would provide adequate data acquisition coverage for direct data transmission during Alaska and CONUS operations and that these stations are, therefore, acceptable for all candidate mission altitudes at and above 366 n mi.

International coverage by EOS requires either (1) large onboard data storage capacity, awaiting data dump over CONUS, (2) a more extensive network of data acquisition centers, or (3) a system of relay satellites which provide continuous or near continuous intermediary contact between EOS and the designated data acquisition ground stations. The effect of utilizing the TDRS system for this function was studied; the results, along with a map of the zone of exclusion, are presented in Report No. 3, Subsection 6.12.

6 - LAUNCH VEHICLE PERFORMANCE

This section presents candidate launch vehicle performance and mission scenario descriptions for initial deployment of the EOS family of spacecraft using conventional boosters. Also discussed are the performance considerations of using the Shuttle to perform the deployments with the added capability of retrieval and resupply. Shuttle discussions include dual EOS deployment/service performance considerations.

Also included are detailed weight summaries of the family of EOS spacecraft, i.e., EOS-A, B, C, D, E, F, SEASAT-A and the Solar Maximum Mission (SMM). Conflicts between the basic EOS design and the follow-on requirements are also addressed.

6.1 CANDIDATE LAUNCH VEHICLES

6.1.1 CONVENTIONAL LAUNCH VEHICLE CANDIDATES

Initial deployment of the EOS class of satellites can be accomplished using four types of conventional launch vehicles. Table 6-1 summarizes the EOS mission orbit, launch vehicle and maximum deployment capability, and approximate launch date for each mission.

The Delta 2910, Delta 3910, and Titan III B (SSB) are used to deliver EOS satellites which have sun-synchronous and polar mission orbits. The performance of each of these launch vehicles is shown in Fig. 6-1 for sun-synchronous orbits (EOS missions A, B, C and E), and Figure 6-2 for the polar mission (EOS-D). Each figure also illustrates the projected weight of each satellite in comparison to launch vehicle capability. Figure 6-3 presents the same parameters for the Titan III C7 launch vehicle which is used for

Table 6-1 Candidate Conventional Boosters For EOS and Follow-on Missions

EOS MISSION	INITIAL LAUNCH YEAR	MISSION ORBIT, N MI	INCLINATION, DEG	WEIGHT WITH AKM, LB	CANDIDATE BOOSTER	LAUNCH SITE	PAYLOAD MARGIN, LB
A	'79	366.1 x 366.1	98.1	2401	DELTA 2910	WTR	259
B	'81	366.1 x 366.1	98.1	2837	DELTA 3910	WTR	893
C	'80	366.1 x 366.1	98.1	4743	TITAN III B/NUS*	WTR	407
D	'81	324.2 x 324.2	90.0	2820	DELTA 2910	WTR	5
E	'82	450 x 450	98.7	3481	DELTA 3910	WTR	69
F	'81	19323 x 19323	0	4360	TITAN III C7	ETR	340
SEASAT-A	'78	432 x 432	108.0	2809	DELTA 3910	WTR	541
SMM	'78	300 x 300	28.0	3538	DELTA 2910	ETR	362

T1-7

*SRM USED FOR CIRCULARIZATION AT MISSION ALTITUDE the EOS-F mission to geosynchronous equatorial (19,323 n mi altitude, 0 deg inclination) orbit. The Titan III C7, which has a Transtage as an upper stage, places the EOS-F into a geosynchronous transfer ellipse. Since the Transtage has propellant remaining after it performs the transfer ellipse maneuvers, it is retained to perform the circularization maneuver at apogee (19,323 n mi).

6.1.2 UNAUGMENTED SINGLE DEPLOYMENT AND RESUPPLY USING SHUTTLE

The Shuttle was eliminated as a viable booster for the initial launch of the vehicles for several reasons. Shuttle launch facilities at WTR will not become available until 1983, which is too late if the EOS traffic model (refer to Table 6-1) were adhered to. Secondly, the Shuttle facilities at ETR will not be available until the latter part of 1979, thus eliminating the Shuttle as a booster for the Solar Maximum Mission (SMM). Attempts to capture the EOS and follow-on traffic by using the Shuttle/OOS from ETR during the 1979-1982 time frame have been made, and the results are presented in Fig. 6-4. The large plane change maneuvers imposed on the OOS (assumed to be a Transtage vehicle) raised the ΔV requirements beyond the capability of the Shuttle/OOS except in one instance, the EOS follow-on mission D. Even though the mission is captured it is not considered a viable option since it requires expending a OOS vehicle in the process of reaching its mission orbit.

After the initial launch of EOS spacecraft by conventional launch vehicles, the Shuttle can be used to deploy additional vehicles (deploy), replace vehicles (deploy-retrieve or round-trip), and to resupply or service them. Figure 6-5 presents the Shuttle payload capability to sun-synchronous altitudes and inclinations, and compares the payload requirements of the EOS-A, -B, -C, and -E missions to Shuttle capability. Deployment into the mission orbits without using kick stages can be accomplished for the EOS-A, -B, and -C missions. Resupply of these vehicles has been considered and the payload requirements for a resupply (and possible retrieval) mission are also presented in Fig. 6-5. The EOS-A, -B, and -C spacecraft can be resupplied directly by the Shuttle in their mission orbits. Resupply of EOS-E in its sun-synchronous mission orbit is beyond Shuttle capability, and provisions must be made for resupply (or servicing) in a lower altitude orbit. The EOS-E spacecraft cannot be deployed directly into its 450-n mi mission orbit by the Shuttle; it requires the assistance of kick stages to get into the mission orbit, and later, kick stages to return to the Shuttle for servicing. Deployment and resupply of EOS-E will be discussed in more detail in Subsection 6.1.3.

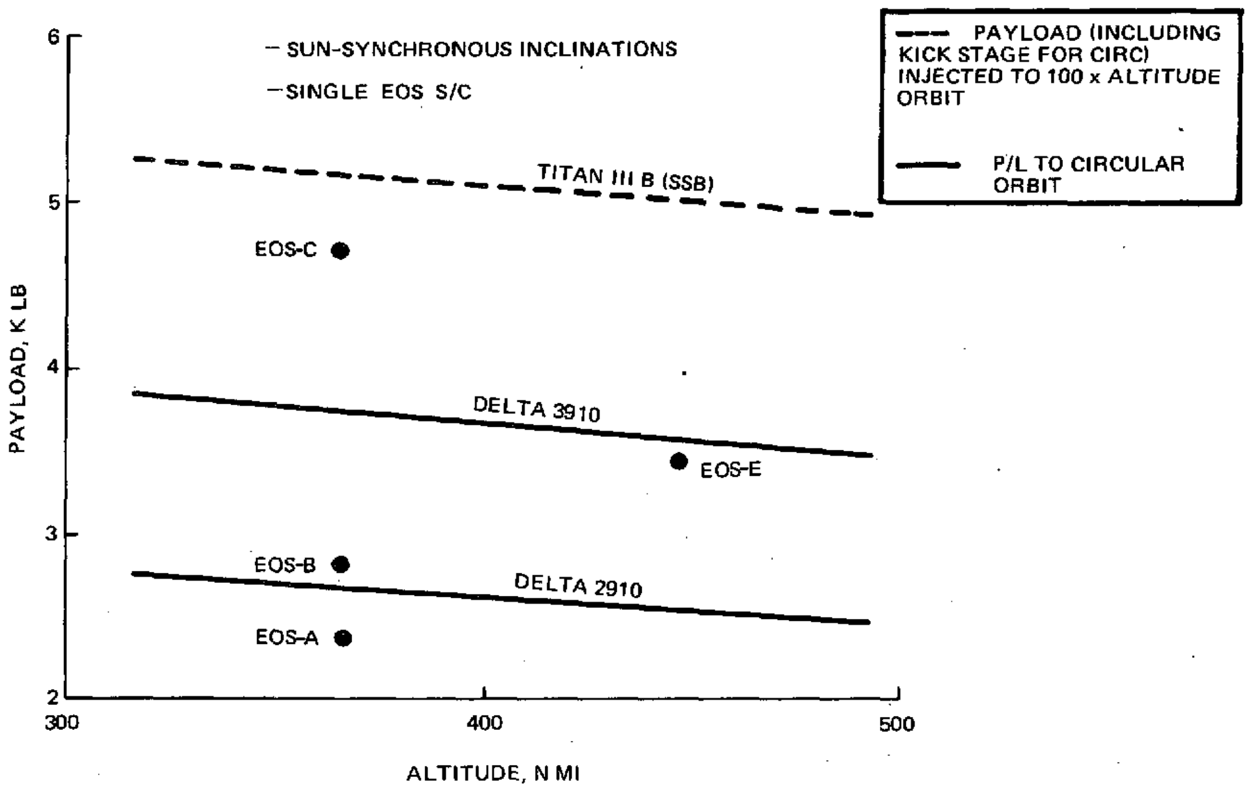

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Fig. 6-1 Conventional Launch Vehicle Capability

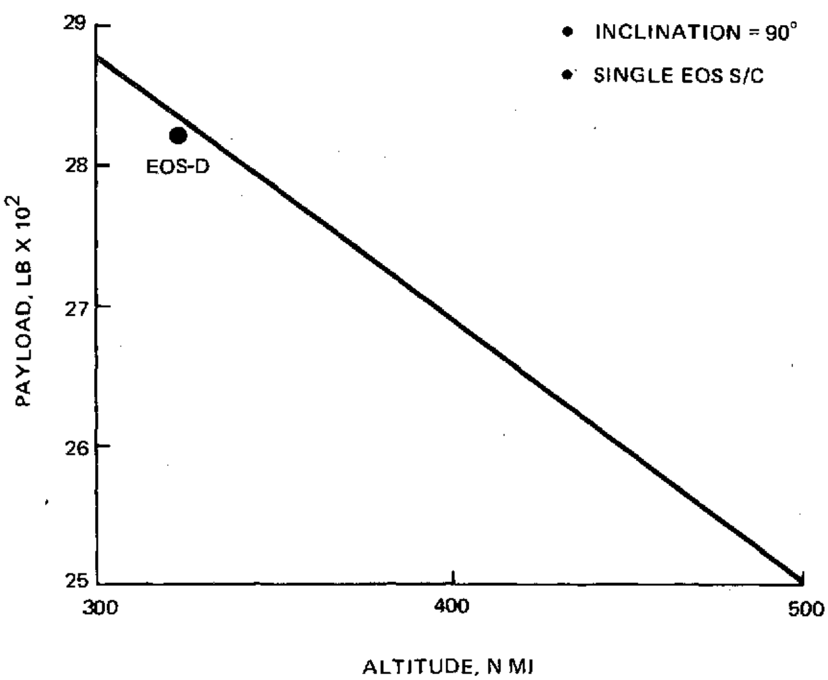

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Fig. 6-2 Delta 2910 Capability to Polar Orbits

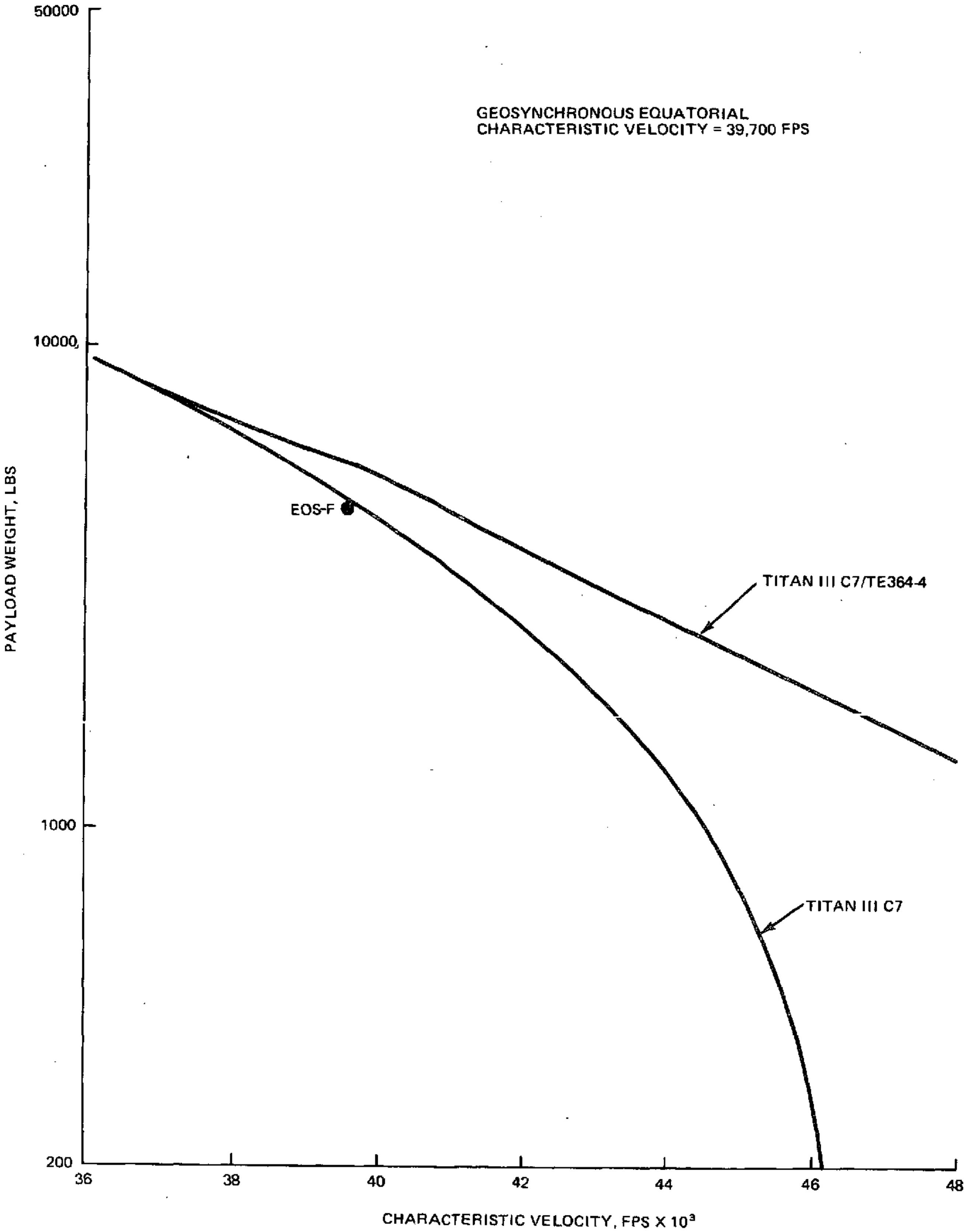

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Fig. 6-3 Titan III C7 Payload Weight Vs Characteristic Velocity – ETR Launch

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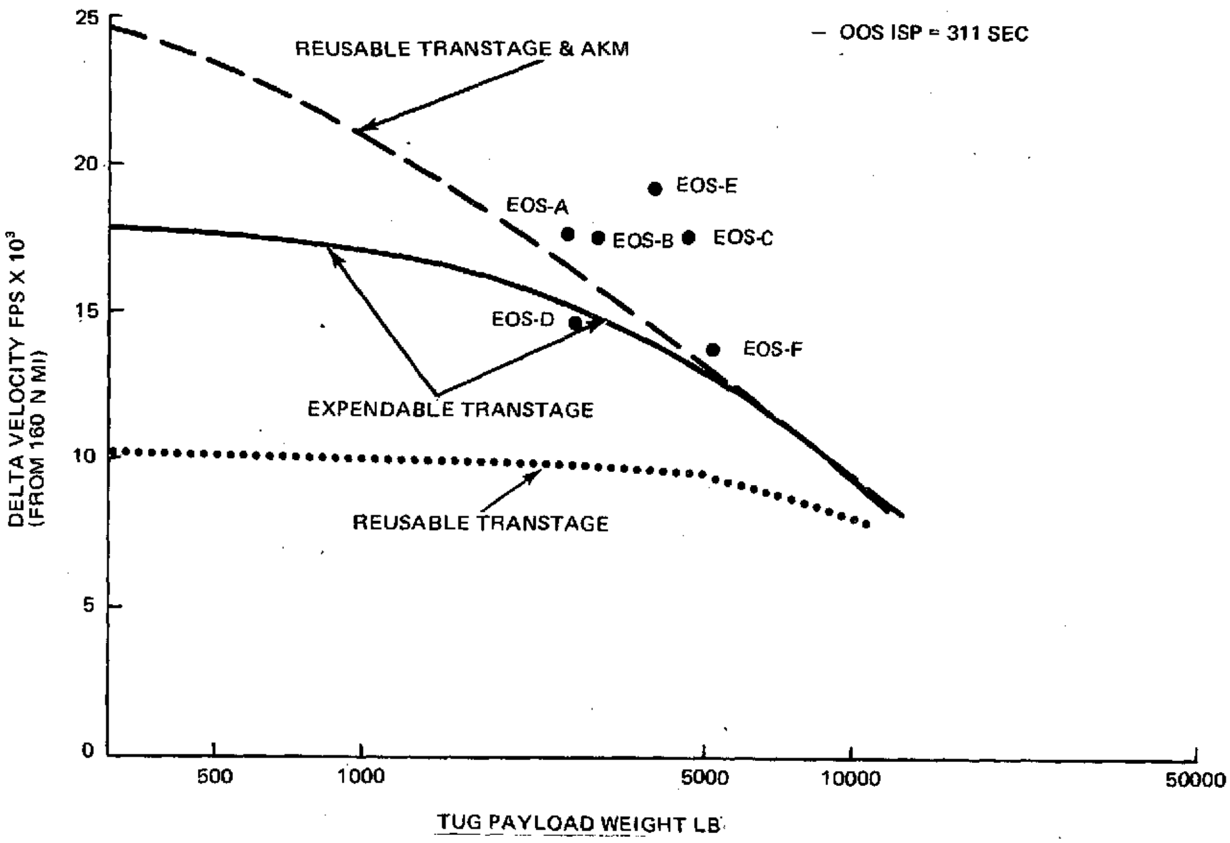
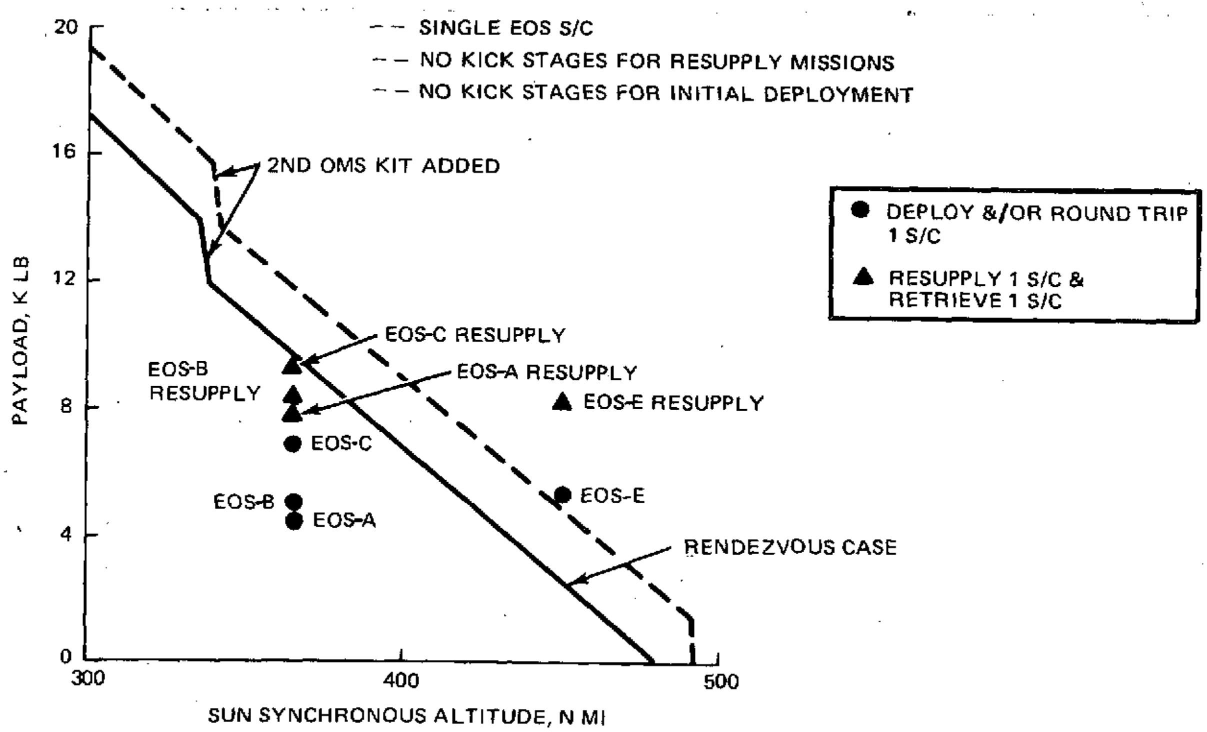


1-24 **Fig. 6-4 ETR Shuttle Launch to 160 N Mi and Transtage (OOS) to Mission Orbit**

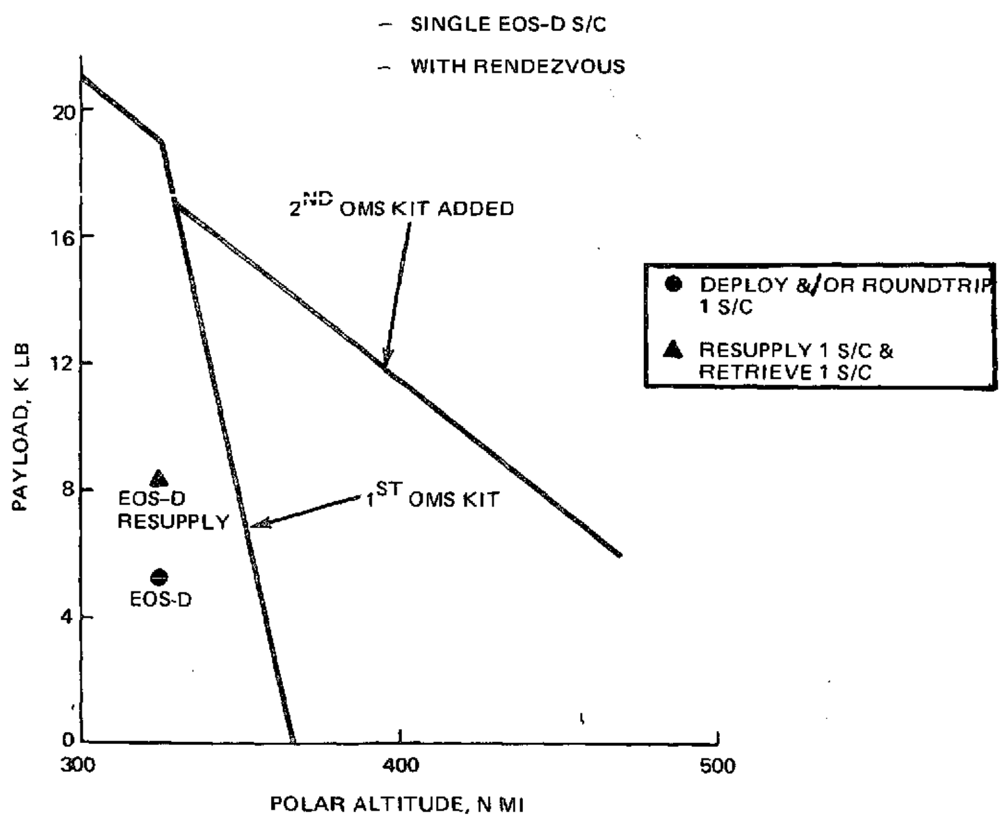


6-20 **Fig. 6-5 Shuttle Payload Capability to Sun-Synchronous Altitudes**

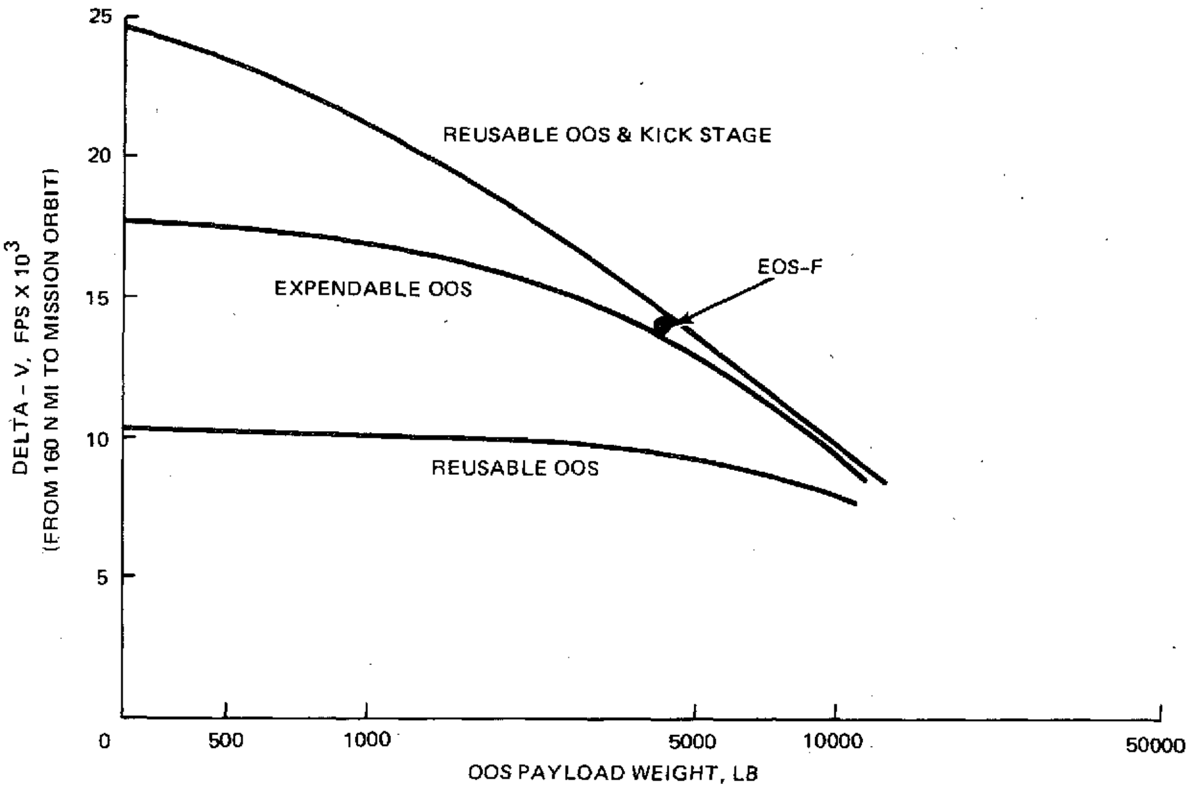
Figure 6-6 illustrates that EOS-D can be deployed and resupplied in its mission orbit of 324 n mi at 90 deg inclination. In this instance, the Shuttle capability with one OMS kit aboard far exceeds the EOS-D requirement; thus there is room for payload growth without affecting delivery or resupply capability.

6.1.3 AUGMENTED SINGLE DEPLOYMENT AND RESUPPLY USING THE SHUTTLE

The EOS-F mission orbit requirements far exceed the unassisted Shuttle capability and require the use of either the Orbit-to-Orbit Shuttle (OOS) or Shuttle/Tug to reach its geosynchronous equatorial mission orbit. The OOS is envisioned as an adaptation of an existing stage which is scheduled to become operational in 1979 and remain so until 1983, when a newly developed Tug is scheduled to become operational. Figure 6-7 shows the EOS-F performance requirement (4200 lb and 14,000 fps) and the deploy capability of an OOS (a derivative of the Transtage) operating in several modes from the Shuttle (160 n mi parking orbit). If the OOS is to be recovered, it will release EOS-F in a 160 x 19,323-n mi orbit and a kick stage must be used to circularize the EOS-F at geosynchronous altitude.



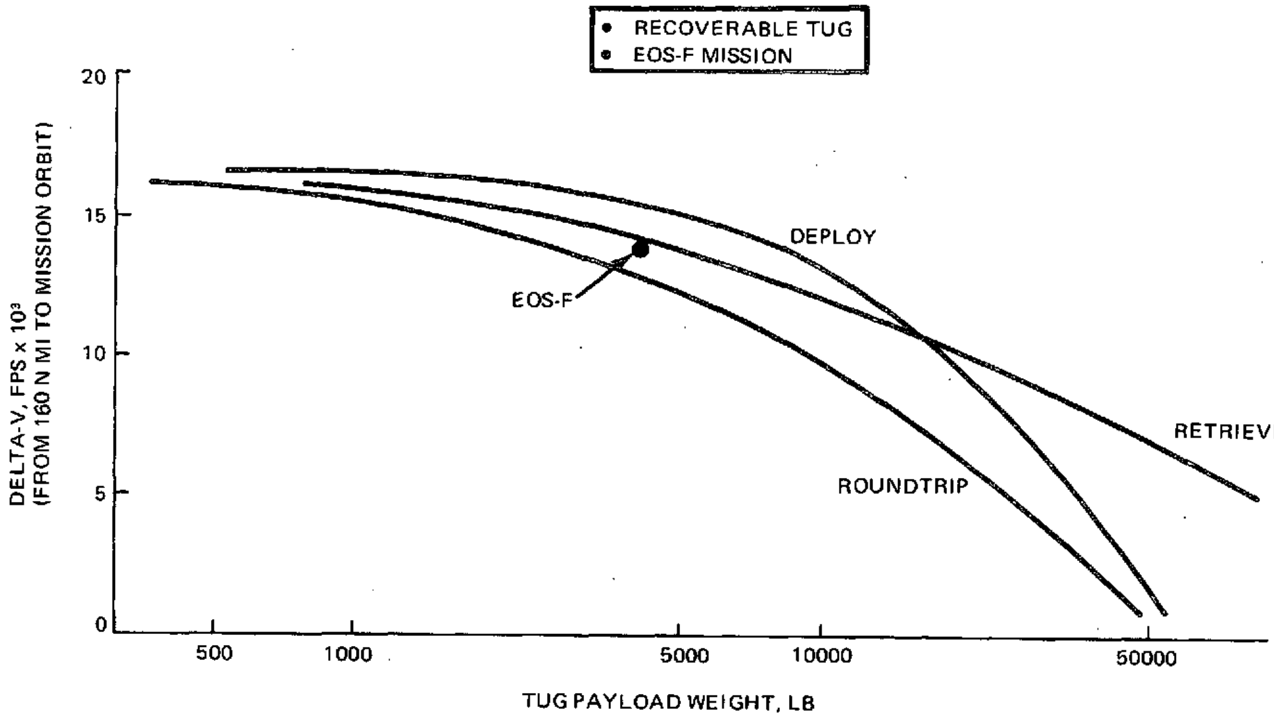
6-21 Fig. 6-6 Shuttle Capability to Polar Orbits



6-22 **Fig. 6-7 Shuttle/OOS (Transtage) Deploy Performance From 160 Nautical Miles**

The Shuttle-carried resupply weight for EOS-F is approximately 4000 lb, comprised of the FSS, SPMS, and EOS replacement modules. This would be considered a roundtrip payload on an OOS or Tug since modules which are brought to the EOS would be exchanged for units of equal weight. The capability of the OOS falls short of this resupply requirement; thus, resupply or retrieval can only be considered when the full capability Tug becomes operational in 1983. Figure 6-8 shows that resupply of the EOS-F in geosynchronous equatorial orbit (ΔV = 14,000 fps) is beyond the capability of even the full-capability Tug. An alternative (not considering cost) would be to retrieve the EOS-F with the Tug and return it to earth for refurbishment. The figure shows that the Tug does have the capability to deploy (or retrieve) the EOS-F on separate Shuttle flights.

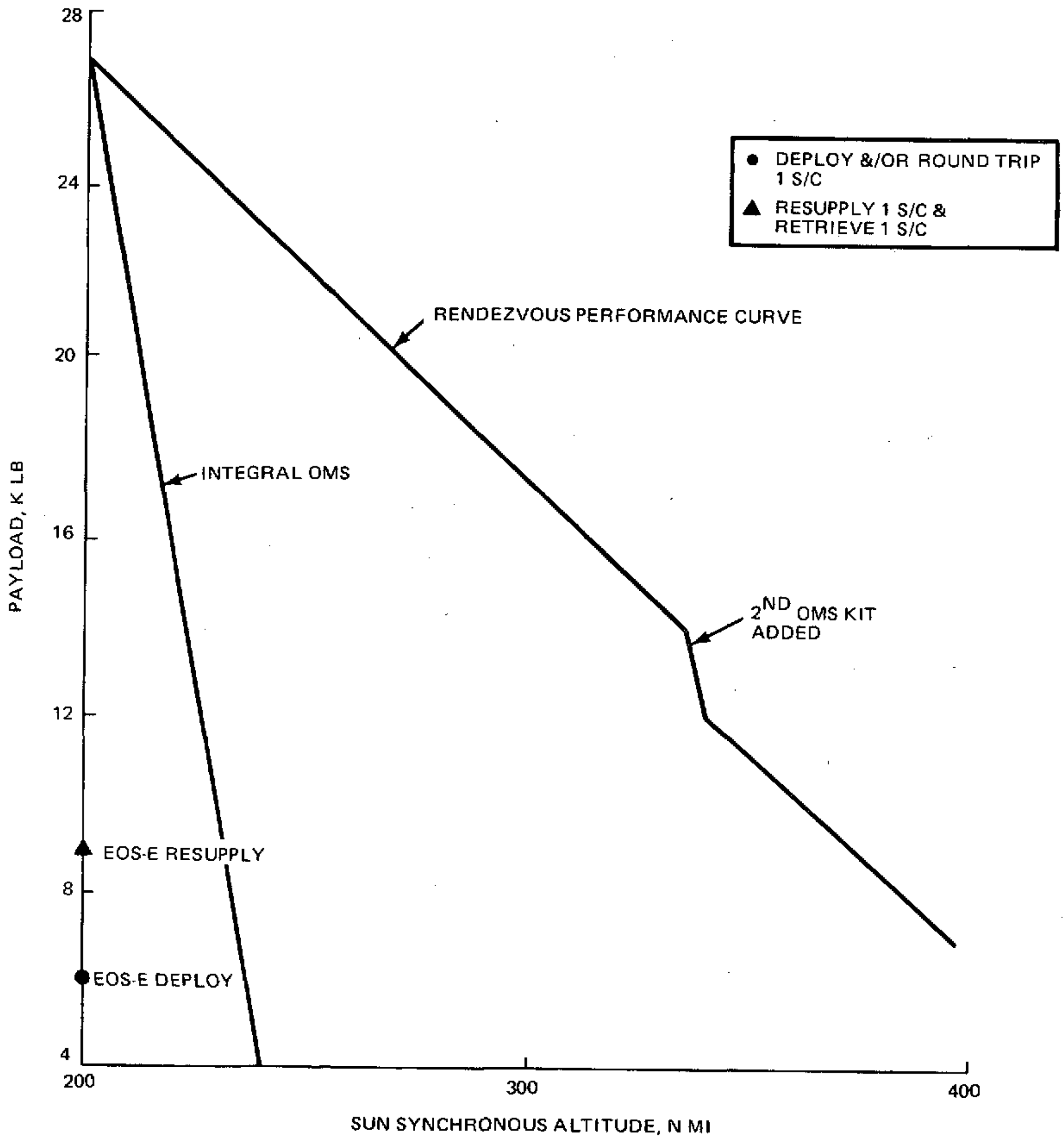
In summary, the Shuttle, in conjunction with either an OOS or a Tug, can deploy the EOS-F vehicle. Shuttle-based resupply using either the OOS or Tug is not possible in the geosynchronous mission orbit. Resupply of EOS-F using a Tug-mounted resupply system cannot be accomplished unless the combined weight of the resupply system and replacement EOS modules is kept below 2700 lb. Retrieval of the vehicle is possible using the Tug.



6-23 **Fig. 6-8 Shuttle/Cryogenic Tug Performance From 160 Nautical Miles**

As mentioned earlier, the EOS-E deployment and resupply missions cannot be accomplished without augmenting Shuttle capability (see Fig. 6-5). To plan for the resupply of EOS-E, the payload must be outfitted with a 4-SRM kick stage and deployed by the Shuttle at a 200-n mi circular orbit. The kick stage would then be used to attain the 450-n mi mission orbit. When resupply is required, the kick stage would de-orbit EOS-E by lowering perigee to 200 n mi; after coasting to perigee, the last SRM would circularize the vehicle at 200 n mi which is the Shuttle parking orbit altitude for this case. After EOS-E has been serviced in the low-altitude Shuttle orbit, it would be equipped with a four-SRM kick stage: two SRM's for ascent to its original mission orbit and two for return to the Shuttle for resupply or service. Although resupply or service of the satellite could be performed in the elliptical orbit, preliminary analyses performed by NASA (JSC) indicated that circular orbit servicing is preferable (References 6.9-1 through 6.9-3).

Figure 6-9 indicates that the initial deployment of EOS-E to 200 n mi, and later, resupply in a 200-n mi circular orbit, can be accomplished by the Shuttle using its integral OMS tankage only.



6-24 **Fig. 6-9 Shuttle Performance to Sun-Synchronous Altitudes, EOS-E Deployment - Resupply**

6.1.4 DUAL EOS DEPLOYMENT USING THE SHUTTLE

CIRCULAR ORBIT DEPLOYMENT AND RESUPPLY - Dual deployment of EOS-A, B, and C spacecraft to a 200-n mi orbit (SRM to mission orbit) has been analyzed and determined to be within the Shuttle integral OMS capability. Figure 6-10 presents the Shuttle payload capability and payload requirements of EOS dual launches. Dual deployment directly into the mission orbits is beyond Shuttle capability; deployment into elliptical orbits with apogee at the mission orbit altitude is feasible and is discussed later.

Each EOS spacecraft deployed at 200 n mi would have a four-SRM kick stage with the following purposes:

- SRM No. 1 - Initiates transfer from 200-n mi circular to mission orbit
- SRM No. 2 - Circularizes EOS at mission altitude
- SRM No. 3 - Initiates transfer from mission orbit to 200-n mi orbit for service or resupply
- SRM No. 4 - Circularizes EOS at 200 n mi for service or resupply.

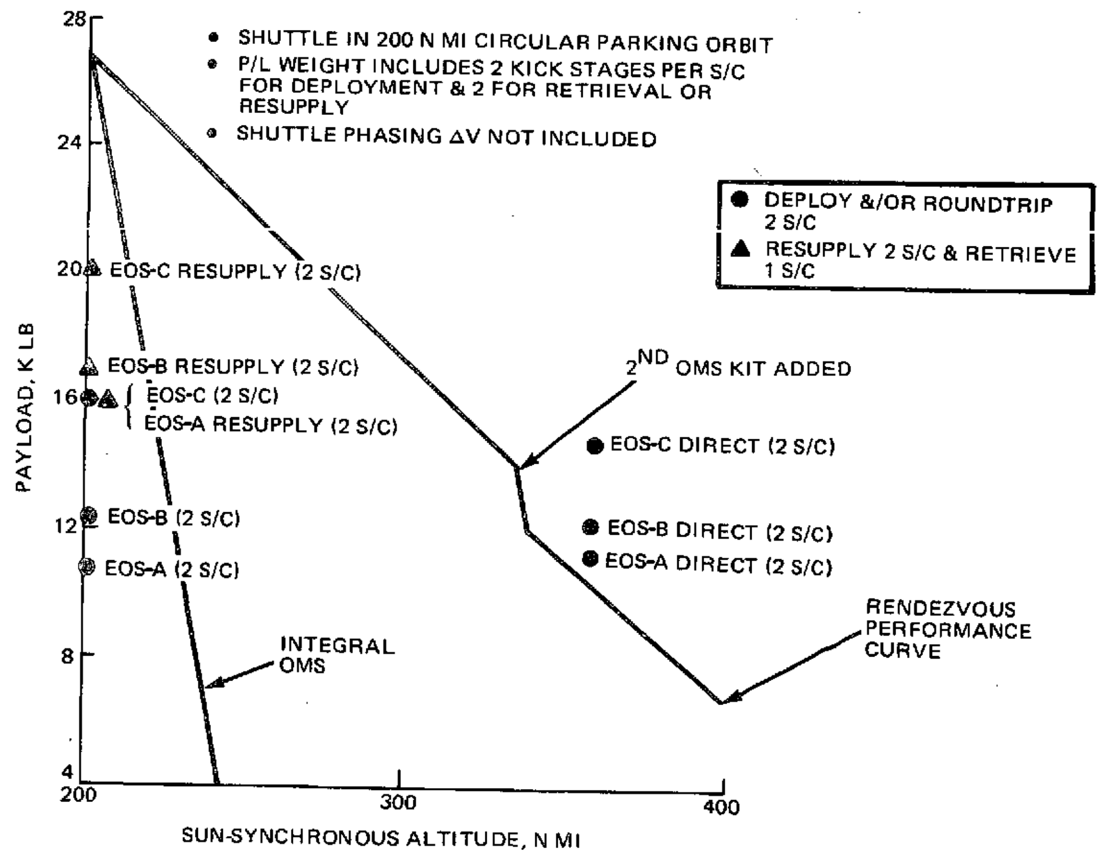


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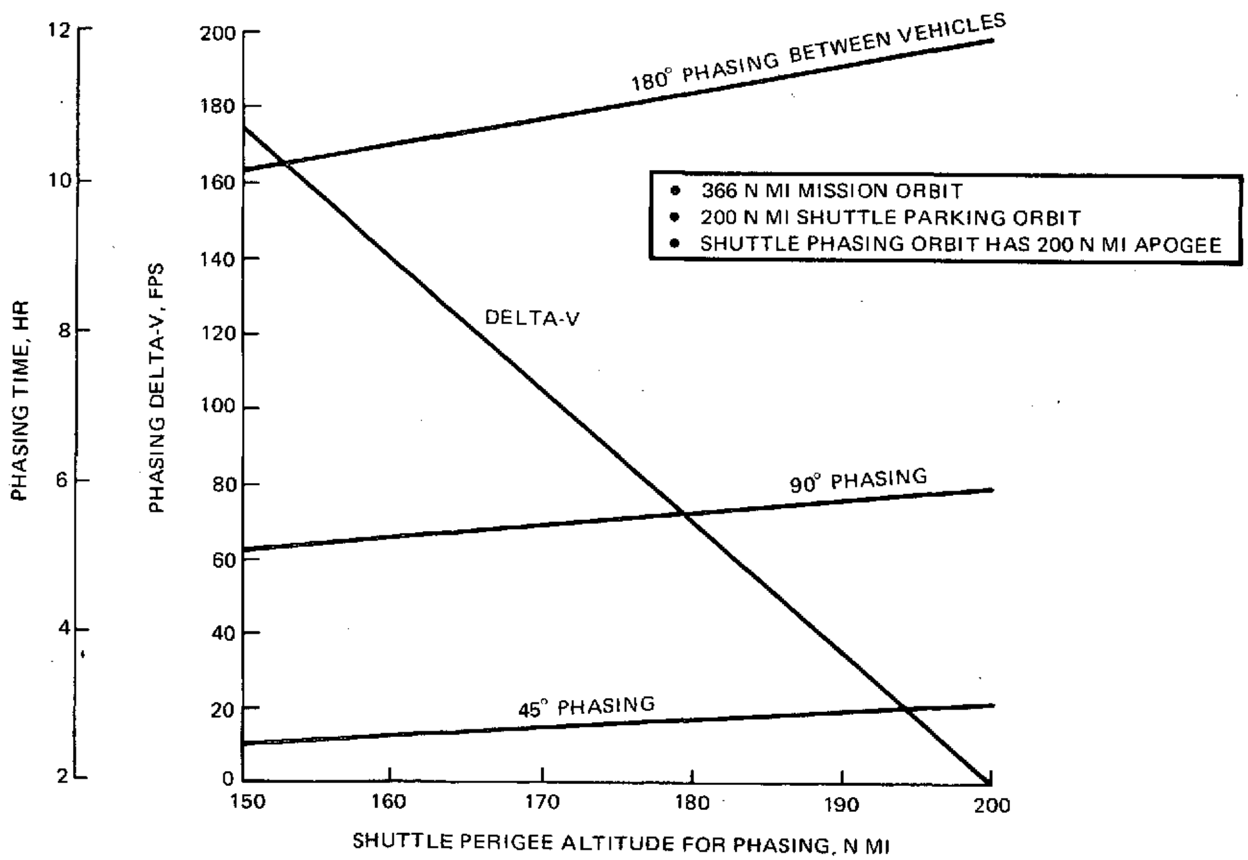
Fig. 6-10 Shuttle Capability to Sun-Synchronous Altitudes

After the first EOS kick stage transfer maneuver at 200 n mi, the Shuttle coasts to set up the proper phasing between vehicles. Figure 6-11 presents phasing delta V and phasing time characteristics for various phasing angles between vehicles in a 366-n mi mission orbit. The Shuttle can remain in its 200-n mi parking orbit and phase with the first deployed EOS without the expenditure of OMS phasing delta V. This phasing results from the difference in the period (and the corresponding angular velocity) of the Shuttle and the deployed EOS vehicle. The Shuttle can also lower its perigee, thereby increasing the mean differential angular motion to make up the required phasing angle in a shorter time. However, Fig. 6-11 indicates that the phasing time saved by lowering perigee is not worth the expenditure of the additional delta V required, and that the gross phasing should be performed in the 200 n mi parking orbit.

Resupply of dual EOS-A, -B and C vehicles would take place in the 200-n mi Shuttle parking orbit; the payload requirements for the resupply are well within Shuttle integral OMS capability (Fig. 6-10). The payload requirements include the return of one EOS spacecraft to the ground.



6-26 **Fig. 6-11 Shuttle Phasing Characteristics for Dual EOS Deployment**

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ELLIPTIC ORBIT DEPLOYMENT AND RESUPPLY - A typical dual EOS elliptical-orbit deployment scenario would begin by Shuttle launch and insertion into a 50x100-n mi orbit at perigee, followed by a coast to 100 n mi at which point an OMS maneuver would produce an apogee at the mission orbit altitude and a perigee at 100 n mi. While in the elliptical transfer orbit, the EOS satellite would be separated; at apogee the EOS circularizes with a kick stage as the Shuttle coasts in the elliptic transfer orbit. Coasting in the elliptic transfer orbit will result in phasing between EOS deployments. Figure 6-12 presents the time that the Shuttle must coast after deploying the first EOS spacecraft to attain 180 deg separation between the first and second EOS. The data is presented for various Shuttle apogee (mission orbit) altitudes. In addition, the delta V that the EOS needs to circularize at the mission orbit altitude from the Shuttle transfer orbit is presented. Also presented is the impulsive delta V that the OMS must supply to get the Shuttle from the 50 x 100- n mi insertion orbit onto the transfer orbit.

Figure 6-13 presents the Shuttle capability to elliptic transfer orbits of various apogee altitude in comparison to the payload requirements of the EOS-A, -B, and -C dual deployment missions, and the EOS-B and -C resupply missions. All of these deployment and resupply missions are within Shuttle capability in operating on the integral OMS tankage.

Resupply of the dual EOS by the Shuttle can be performed in the elliptic transfer orbit, but the analyses documented in Reference 6.9-2 suggest that the circular orbit approach provides "significant advantages over the elliptic orbit in terms of the mission planning cycle."

6.2 SPACECRAFT WEIGHTS

The EOS spacecraft weights for missions A, B, and C, and follow-on missions are shown in Table 6-2. These weights were built up using the Barebones Spacecraft as a base, and adding operational options and structural and subsystem increased capability as dictated by each mission.

Options added to each spacecraft are:

- Retrieval Capability - Structural interface for mating with Orbiter Flight Support System (FSS)
- Resupply Capability - Mechanical and electrical devices required to accomplish on-orbit replacement of modules, RCS and solar array by an SPMS-equipped orbiter (except EOS-A)
- Two-year Service Life - Additional battery to reduce depth of discharge, extending battery life, where required.

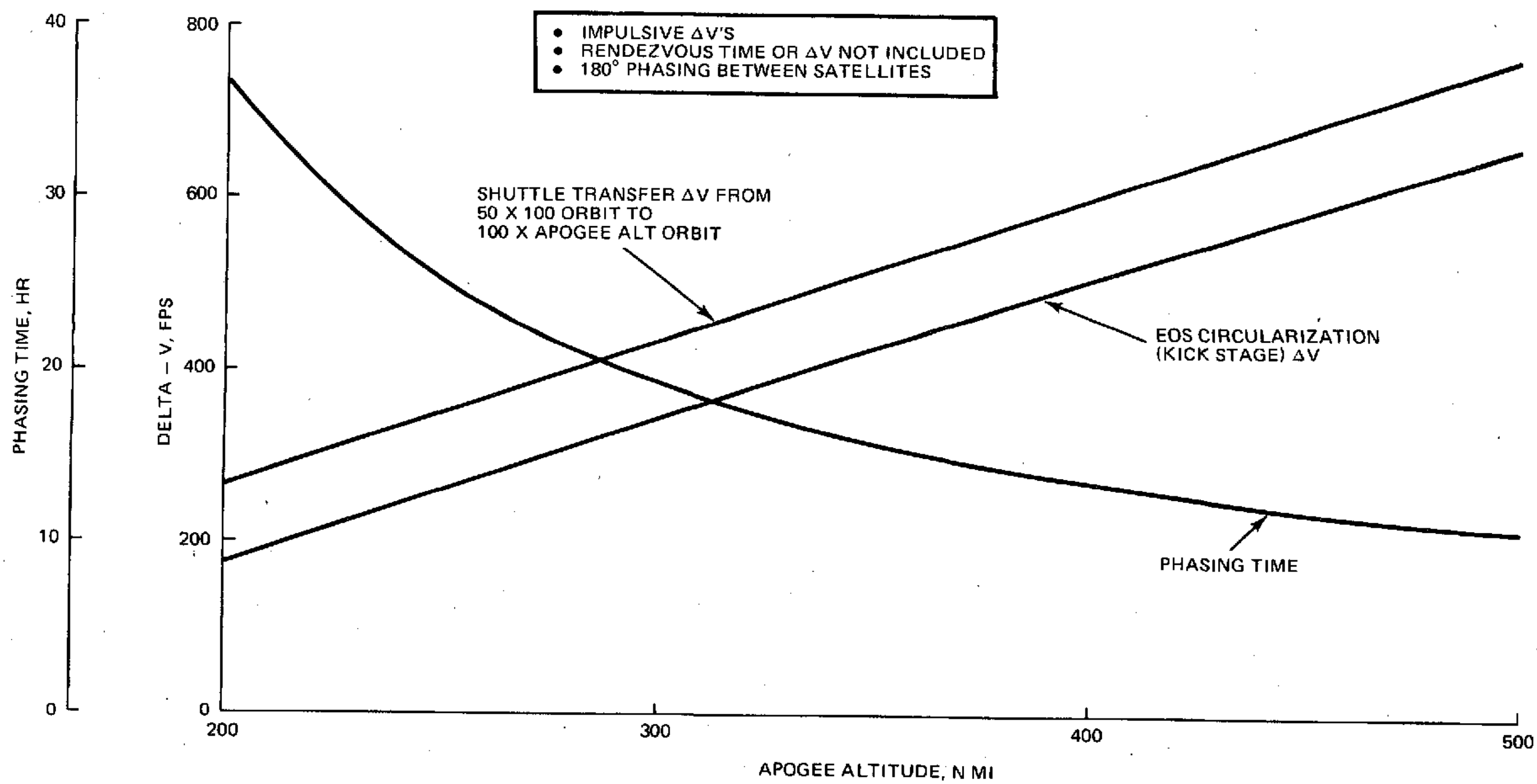
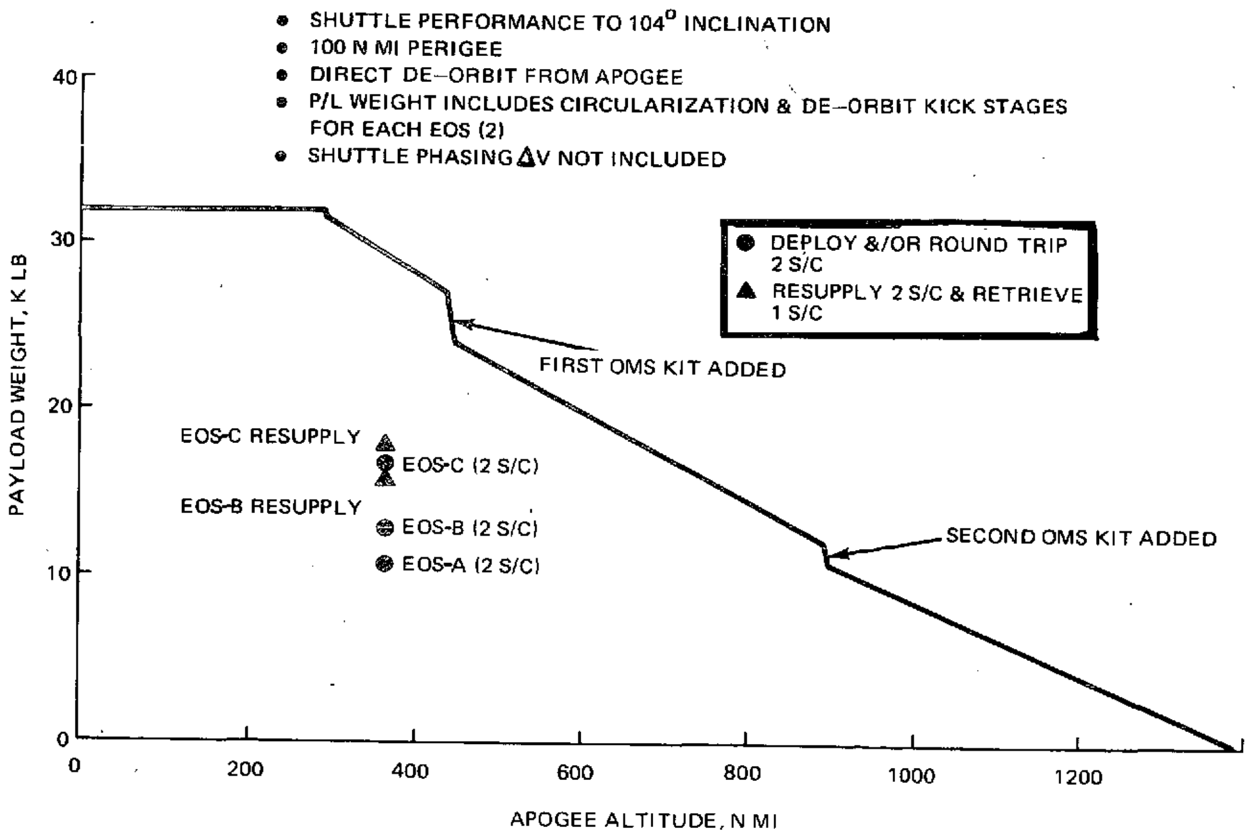


Fig. 6-12 Phasing and Circularization Characteristics of Multiple EOS Deployments Using the Shuttle Elliptical Orbits

In addition to these increases to the basic spacecraft, a weight allowance is created to provide for local reinforcement on the heavier EOS-C and -F spacecraft, and for changes to the launch adapter.

The changes in the basic spacecraft required by specific mission demands are shown as spacecraft mission peculiar items. These include additional batteries, enlarged solar array, larger ACS reaction wheels and torquer bars, memory module, and increased RCS propellant capacity to perform the orbit adjust and thrust vector control functions. For the EOS-C, an SRM is included for circularization at the mission altitude. For EOS-E, a three SRM kick stage is included for circularization at the mission altitude and for lowering the spacecraft to a low parking orbit for retrieval or resupply.

The instrument and mission peculiar group is composed of those items which are required only in support of the instrument payload. This includes support structure, resupply mechanisms, and thermal insulation, which are grouped under instrument support, as well



6-28 Fig. 6-13 Shuttle Payload Weight Versus Elliptical Orbital Altitude, WTR

Table 6-2 EOS and Follow-on Mission Weight and Launch Vehicle Performance

ITEM DESCRIPTION	EOS-A	EOS-B	EOS-C	EOS-D (SEASAT-B)	EOS-E (TIROS-O)	EOS-F (SEOS)	SEASAT-A	SMM
• BAREBONES SPACECRAFT WEIGHT-LB[1]	1361	1361	1361	1361	1361	1361	1361	1361
– ORBITER DEPLOY PENALTY	67	67	67	67	67	64	67	67
– ORBITER RETRIEVAL PENALTY	1	1	1	2	1	1	2	2
– ORBITER RESUPPLY PENALTY	–	115	115	128	115	115	128	128
– 2-YEAR SERVICE LIFE (BATTERY)	–	–	32	32	–	–	32	–
– INCREASED STRUCTURAL CAPABILITY	–	–	60	–	–	80	–	–
– Δ CONTINGENCY	11	21	36	26	21	36	26	23
• BASIC SPACECRAFT	1440	1565	1672	1616	1565	1657	1616	1581
– SPACECRAFT MISSION PECULIAR	(47)	(47)	(629)	(200)	(718)	(59)	(175)	(59)
○ THERMAL CONTROL	–	–	–	60	–	75	60	60
○ SOLAR ARRAY	–	–	84	84	–	-60	61	-75
○ ATTITUDE CONTROL	–	–	145	–	–	6	–	37
○ COMM & DATA HANDLING	18	18	18	18	18	9	18	18
○ ORBIT ADJUST/TRANSFER	27	27	340	27	682	41	27	27
○ Δ CONTINGENCY	2	2	42	11	18	-12	9	-8
– INSTRUMENT MISSION PECULIAR	(354)	(425)	(742)	(431)	(428)	(344)	(431)	(467)
○ INSTRUMENT SUPPORT[2]	136	189	445	235	198	214	235	231
○ TDRSS COMMUNICATION	87	87	87	87	87	–	87	87
○ WIDE BAND COMM & DATA HANDLING	88	96	112	46	88	88	46	88
○ Δ CONTINGENCY	43	53	98	63	55	42	63	61
– INSTRUMENTS	(560)	(800)	(1700)	(706)	(770)	(2300)	(587)	(1431)
○ MULTI-SPECTRAL SCANNER	160	–	–	–	–	–	–	–
○ THEMATIC MAPPER	400	400	800	–	–	–	–	–
○ HIGH-RESOLUTION POINTABLE IMAGER	–	400	400	–	–	–	–	–
○ SYNTHETIC APERTURE RADAR	–	–	500	–	–	–	–	–
○ SEASAT-B (OCEAN DYN & SEA ICE)	–	–	–	706	–	–	–	–
○ TIROS-O (WEATHER & CLIMATE)	–	–	–	–	770	–	–	–
○ SEOS (GEOSYNCHRONOUS EOS)	–	–	–	–	–	2300	–	–
○ OTHER EXPERIMENTS	–	–	–	–	–	–	587	1431
• SUBTOTAL – SPACECRAFT	2401	2837	4743	2953	3481	4360	2809	3538
WEIGHT SAVING OPTIONS[3]	–	–	–	-133	–	–	–	–
• TOTAL SPACECRAFT WEIGHT - LB	2401	2837	4743	2820	3481	4360	2809	3538
• LAUNCH VEHICLE PAYLOAD CAPABILITY	2660	3730	5150	2825	3550	4700	3350	3900
• PAYLOAD MARGIN - LB	259	893	407	5	69	340	541	362
• LAUNCH VEHICLE[4]	D2910	D3910	TIIIB	D2910	D3910	TIIIC-7	D3910	D2910

NOTES: (1) BAREBONES SPACECRAFT WEIGHT INCLUDES 146 LB CONTINGENCY.

(2) INSTRUMENT SUPPORT WEIGHT INCLUDES RETRIEVAL STOWAGE LOCKS AND RESUPPLY MECHANISMS (EXCLUDING EOS-A) FOR IMP AND INSTRUMENTS

(3) WEIGHT SAVING OPTIONS EMPLOYED ARE:

a. ROLL-OUT SOLAR ARRAY (EOS-D) SAVINGS INCLUDE CONTINGENCY REDUCTION.

(4) TIIIB PAYLOAD LIMITS ARE FOR TITAN IIIB (SSB)/NUS.

as support equipment such as wide band (WB) tape recorders and WB communications. Included in WB communications are the MOMS and signal conditioning units.

The instrument group completes the buildup of launch weight. The launch weight is compared to the candidate booster performances; employing various weight reduction options as required to provide a positive weight margin. In the case of EOS-D, it was necessary to use a roll-out (flexible) solar array in place of the rigid deployable solar array, for a savings of 133 lb, including a 10% contingency. Additional margin (36 lb) may be obtained by substituting a hybrid composite instrument truss for the current aluminum design.

The payload weight for EOS-A through -F missions using the Shuttle as the launch vehicle are shown in Tables 6-3 through 6-5. The Flight Support System weight included in the payload weights are detailed in Tables 6-6 through 6-8.

Table 6-3 describes the functional weight breakdown of EOS-A through -F payloads for single spacecraft deploy/retrieve missions. In general, the observatory weight is the same as shown in Table 6-2, except for the deletion of the launch adapter. The exceptions to this are EOS-A (resupply) and those spacecraft which required kick stages: EOS-C and -E. Since the baseline EOS-A contains no resupply provisions, a resupply-compatible configuration has been added to the payload matrix. The kick stage penalty is deleted for EOS-C since the Shuttle places the spacecraft directly into the 366 n mi mission orbit. EOS-E retains a kick stage but the 205-lb circularization SRM is replaced by two 190-lb SRM's, resulting in a 4-SRM Orbit Adjust/Transfer stage which is used to transfer from the low (168 n mi) Shuttle parking orbit to the mission orbit and, at a later date, to lower the orbit to the Shuttle parking orbit for retrieval or resupply. The Orbit Adjust/Transfer stage for a 200-n mi parking orbit is 100 lb lighter, but the non-recurring cost is considerably higher. The Shuttle payload weight includes the Flight Support System (FSS) required to support the observatory during launch, entry, and landing, and to erect the observatory to the deploy position and release it. For a retrieve mission, the FSS docks with the observatory and lowers it into the stowed position for return. FSS weight details for the deploy mission are given in Table 6-6.

Table 6-4 depicts the payload weight for the Shuttle resupply mission for EOS-A through -F. The weights of the individual flight replaceable modules are shown, and include equipment, structure, wiring, thermal control, and resupply and stowage mechanisms.

Table 6-3 Shuttle Payload Summary - Deploy/Retrieve Mission

FUNCTION	WEIGHT, LB						
	EOS-A BASELINE	EOS-A RESUPPLY	EOS-B	EOS-C	EOS-D SEASAT-B	EOS-E TIROS-O	EOS-F SEOS
• BASIC STRUCTURE	388	440	440	460	450	440	477
• ELECTRICAL POWER	169	169	169	201	201	169	169
• ELECTRICAL HARNESS	45	90	90	90	94	90	90
• SOLAR ARRAY & DRIVE	195	195	195	279	279	195	135
• ATTITUDE CONTROL	161	161	161	306	161	161	167
• RCS (HYDRAZINE)	40	40	40	40	40	40	54
• COMM & DATA HANDLING	146	146	146	146	146	146	137
• THERMAL CONTROL	62	80	80	80	140	80	155
SPACECRAFT, LB	*1206*	*1321*	*1321*	*1602*	*1511*	*1321*	*1384*
• MISSION PECULIAR	(338)	(391)	(399)	(687)	(395)	(1259)	(329)
– ORBIT/ADJUST/TRANSFER	27	27	27	43	27	886	27(2)
– INSTRUMENT SUPPORT	136	189	189	445	235	198	214
– TDRSS COMMUNICATION	87	87	87	87	87	87	–
– WB COMM & DATA HNDLG	88	88	96	112	46	88	88
• INSTRUMENTS	560	560	800	1700	706	770	2300
• CONTINGENCY	202	222	222	322	246	240	212
OBSERVATORY, LB	*2306*	*2494*	*2742*	*4311*	*2858*	*3590*	*4225*
• FLIGHT SUPPORT SYSTEM	2528	2528	2528	2528	2528	2528	200(1)
SHUTTLE PAYLOAD, LB	*4834*	*5022*	*5270*	*6839*	*5386*	*6118*	*4425*

NOTES: (1) FLIGHT SUPPORT SYSTEM FOR EOS-F CONSISTS OF EOS/TUG ADAPTER; PAYLOAD WEIGHT IS USED FOR COMPARISON TO TUG PERFORMANCE.

(2) KICK STAGES FOR 168 N MI PARKING ORBIT; FOR 200 N MI PARKING ORBIT SUBTRACT 100 LB.

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Table 6-4 Shuttle Payload Summary - Resupply Mission

FLIGHT REPLACEABLE MODULES	WEIGHT, LB					
	EOS-A RESUPPLY	EOS-B	EOS-C	EOS-D SEASAT-B	EOS-E TIROS-O	EOS-F SEOS
• ELECTRICAL POWER	256	256	288	288	256	256
• SOLAR ARRAY & DRIVE	207	207	291	292	207	147
• ATTITUDE CONTROL	241	241	386	241	241	247
• RCS/ORBIT ADJUST/TRANSFER	111	111	127	111	970 (2)	125
• COMM & DATA HNDLG	229	229	229	229	229	220
SPACECRAFT MODULES, LB	*1044*	*1044*	*1321*	*1161*	*1903*	*995*
• INSTRUMENT MISSION PECULIAR BOX	123	131	147	81	123	123
• KU-BAND ANTENNA (TDRSS)	96	96	96	96	96	–
• X-BAND ANTENNA (2)	27	27	27	27	27	27
IMP MODULES, LB	*246*	*254*	*270*	*204*	*246*	*150*
• MULTI-SPECTRAL SCANNER	167	–	–	–	–	–
• THEMATIC MAPPER	407	407	407	–	–	–
• HRPI	–	407	814	–	–	–
• SYNTHETIC APERTURE RADAR	–	–	507	–	–	–
• SEASAT-B (5 MODULES)	–	–	–	750	–	–
• TIROS-O (8 MODULES)	–	–	–	–	832	–
• SEOS (3 MODULES)	–	–	–	–	–	2328
INSTRUMENT MODULES, LB	*574*	*814*	*1728*	*750*	*832*	*2328*
• TOTAL RESUPPLY	1864	2112	3319	2115	2981	3473
• FLIGHT SUPPORT SYSTEM	6035	6035	6035	6035	6035	550 (1)
• CONTINGENCY	146	146	170	160	162	114
SHUTTLE PAYLOAD, LB	*8045*	*8293*	*9524*	*8310*	*9178*	*4137*

NOTES: (1) FLIGHT SUPPORT SYSTEM FOR EOS IS EOS/TUG ADAPTER AND MODULE MANIPULATOR AND STOWAGE SYSTEM. PAYLOAD WEIGHT IS FOR COMPARISON TO TUG PERFORMANCE.

(2) KICK STAGE FOR 168-N MI PARKING ORBIT; FOR 200-N MI PARKING ORBIT SUBTRACT 100 LB.

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Table 6-5 Shuttle Payload Summary - Dual Spacecraft Missions

CONFIGURATION	WEIGHT, LB			
	EOS-A (BASELINE)	EOS-A (RESUPPLY)	EOS-B	EOS-C
DEPLOY MISSION				
○ SPACECRAFT WEIGHT - SINGLE	2,306	2,494	2,742	4,311
○ ADD KICK STAGE PENALTY	396	426	561	931
SPACECRAFT WEIGHT - DUAL	*2,702*	*2,920*	*3,303*	*5,242*
○ TOTAL SPACECRAFT WEIGHT - DUAL	5,404	5,840	6,606	10,484
○ TOTAL FLIGHT SUPPORT SYSTEM	5,413	5,413	5,413	5,413
ORBITER PAYLOAD - DUAL DEPLOY	*10,817*	*11,253*	*12,019*	*15,897*
RESUPPLY MISSION				
○ SPARES WEIGHT - SINGLE	–	2,004	2,252	3,494
○ ADD KICK STAGE PENALTY	–	453	588	974
SPARES WEIGHT - DUAL	–	*2,457*	*2,840*	*4,468*
● TOTAL SPARES WEIGHT - DUAL	–	4,914	5,680	8,936
○ TOTAL FLIGHT SUPPORT SYSTEM	–	11,116	11,116	11,116
ORBITER PAYLOAD - DUAL RESUPPLY		*16,030*	*16,796*	*20,052*

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Table 6-6 Flight Support System Weight for Single Spacecraft Deploy/Retrieve

ITEM	WEIGHT LB
● PAYLOAD RETENTION & POSITIONING SYSTEM	(2367)
– RETENTION CRADLE (RETENTION MECH)	624
– POSITIONING PLATFORM (DEPLOYMENT/DOCKING MECH)	1433
– DATA MGMT, ELECTRICAL, THERMAL	310
● LOAD RETENTION PLATES	(656)
– RETENTION CRADLE	328
– POSITIONING PLATFORM	328
● LESS: PAYLOAD RETENTION ALLOWANCE	495
○ TOTAL	2528

NOTE: PAYLOAD DEPLOYMENT AND RETRIEVAL MECHANISM (PDRM) WEIGHT OF 730 LB IS INCLUDED IN THE ORBITER WEIGHT. IF A SECOND PDRM IS REQUIRED, THE WEIGHT IS CHARGED TO THE PAYLOAD.

T1-23

Table 6-7 Flight Support System Weight for Single Spacecraft Resupply

ITEM	WEIGHT, LB
● PAYLOAD RETENTION & POSITIONING SYSTEM	(2542)
– RETENTION FRAME (UNIQUE ASSY. FIXTURE)	175
– RETENTION CRADLE (RETENTION MECH)	624
– POSITIONING PLATFORM (DEPLOYMENT/DOCKING MECH)	1433
– DATA MGMT, ELECTRICAL, THERMAL	310
● SPECIAL PURPOSE MANIPULATOR SYSTEM	(2840)
– MODULE EXCHANGE MECHANISM	1265
– MODULE MAGAZINE	1160
– MODULE MAGAZINE SUPPORT STRUCTURE	415
● LOAD REACTION PLATES	(1148)
– RETENTION FRAME	164
– RETENTION CRADLE	328
– POSITIONING PLATFORM	328
– SPECIAL PURPOSE MANIPULATOR	328
● LESS: PAYLOAD RETENTION ALLOWANCE	495
● TOTAL	6035

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Table 6-8 Dual Spacecraft Flight Support System

ITEM	DEPLOY – RETRIEVE, LB	RESUPPLY MISSION, LB
• SPECIAL PURPOSE MANIPULATOR SYSTEM	(–)	(2,840)
– MODULE EXCHANGE MECHANISM	–	1,265
– MODULE MAGAZINE	–	1,160
– MODULE MAGAZINE SUPPORT	–	415
• P/L RETENTION & POSITIONING SYSTEM	(2,229)	(1,585)
– RETENTION FRAME	–	–
– RETENTION CRADLE	624	–
– POSITIONING PLATFORM	1,433	1,433
– ELECTRICAL & THERMAL	172	152
• LOAD REACTION PLATES	656	(656)
– RETENTION FRAME	–	–
– RETENTION CRADLE	328	–
– POSITIONING PLATFORM	328	328
– SPMS	–	328
• FLIGHT SUPPORT SYSTEM NO. 2	2,885	5,081
• FLIGHT SUPPORT SYSTEM NO. 1	2,528	6,035
• TOTAL ORBITER FSS	5,413	11,116

NOTE: RESUPPLY MISSION PROVIDES FOR CONTINGENCY RETRIEVAL OF ONE OF THE TWO SERVICED SPACECRAFT IN THE EVENT THAT IT MUST BE RETURNED TO EARTH FOR REPAIR.

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The FSS for this mission includes the deploy mission FSS plus a module magazine for spares stowage, an exchange mechanism, and a retention frame for the stowage of out-size spares such as the solar array, TDRSS antenna package and Synthetic Aperture Radar. Resupply FSS weight details are shown in Table 6-6.

Table 6-5 summarizes the payload weights for dual spacecraft deploy/retrieve and dual spacecraft resupply missions for EOS-A, -B and -C. EOS-A is shown in both the baseline non-resupply and the optional resupply-compatible configuration for the deploy/retrieve missions. Note that for dual missions, it is necessary to add kick stages to all three spacecraft for both deploy/retrieve and resupply missions. Kick stage weights for the various spacecraft and missions are shown in Table 6-9; FSS details for dual missions are shown in Table 6-8.

6.3 DESCRIPTION OF FOLLOW-ON MISSIONS

The EOS follow-on missions include:

- EOS-D (SEASAT-B)
- EOS-E (TIROS-O)
- EOS-F (SEOS)
- SEASAT A
- SSM.

Table 6-9 Solid Rocket Motor (SRM) Weights for EOS Mission Model

• LAUNCH VEHICLE – EOS MISSION o MODE	SRM WEIGHT, LB: ASCENT TRANSFER	SRM WEIGHT, LB: ASCENT CIRC	SRM WEIGHT, LB: DESCENT TRANSFER	SRM WEIGHT, LB: DESCENT CIRC
• TITAN III B (SSB)/NUS				
– EOS-C (1 S/C) DEPLOY	–	256	–	–
– EOS-E (1 S/C) DEPLOY	–	205	177 (1)	170 (1)
• SPACE SHUTTLE				
– CIRCULAR ORBIT				
o EOS-E (1 S/C) DEP/RES	196 (1)	186 (1)	177 (1)	170 (1)
o EOS-E (1 S/C) DEP/RES	172	163	155	150
o EOS-A (2 S/C) DEP/RES	97	93	90	88
o EOS-B (2 S/C) DEP/RES	127	124	118	116
o EOS-C (2 S/C) DEP/RES	211	203	197	193
– ELLIPTICAL ORBIT				
o EOS-A (2 S/C) DEP/RES	–	154	146	–
o EOS-B (2 S/C) DEP/RES	–	190	181	–
o EOS-C (2 S/C) DEP/RES	–	335	319	–

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NOTES:
(1) SRM WEIGHTS SHOWN HERE FOR EOS-E ARE FOR 168 N MI PARKING ORBIT.
(2) UNLESS OTHERWISE NOTED, SRM WEIGHTS ARE FOR 200 N MI PARKING ORBIT.
(3) TOTAL KICK STAGE PENALTIES, INCLUDING SRM, STRUCTURE, TVC PROPELLANT AND PROPULSION ARE 16% GREATER THAN SRM WEIGHTS.
(4) FOR DUAL MISSIONS, SRM WEIGHTS PER SPACECRAFT (S/C) ARE SHOWN.

Included in the following subparagraphs are the mission objectives, a mission description, and a tabular listing of mission equipment for each of these follow-on vehicles.

6.3.1 EOS-D (SEASAT B)

MISSION OBJECTIVES - Provide data for short-wavelength gravity field determination for earthquake and geoid mapping. Provide data in support of ocean studies such as large amplitude ocean features, currents, circulation systems, temporal variations, ocean geoid and surface conditions. These conditions include sea state/surface wave height, wind fields, shelf tides, ocean tides, barometric pressure, storm surges, and tsunamis.

MISSION DESCRIPTION - Nominal circular orbital altitude of 324 n mi (600 km) at an inclination of 90 deg.

MISSION EQUIPMENT - Mission equipment (ref: MSFC Payload Description, Level B, October 1973) is as follows:

- Altimeter (K-band pulsed altimeter)
- Scatterometer (K-band, cw, scanned)

- IR Scanner (thermal channel scanning radiometer)
- Transponder (C-band, satellite-to-satellite)
- Retroreflector (optical quality glass reflectors)
- Transponder (S/C-band, satellite-to-satellite)
- Coherent-Radar Experiments (dual frequency, 155 and 1215 MHz altimeter).

6.3.2 EOS-E (TIROS O)

MISSION OBJECTIVES - The TIROS O vehicle is intended to verify for operational use an advanced environmental operation payload. This spacecraft will have implemented operational versions of remote sensing techniques proven in Nimbus and EOS flight experiments as well as improvements in those sensors carried by the previous N/ITOS vehicle. The TIROS O satellite will be the first of the operational vehicles to be designed with the Shuttle exploitive modular design so that in-orbit refurbishment of the payload can be effected and evaluated.

MISSION DESCRIPTION - Nominal altitude of 450 n mi (833 km) circular at an inclination of 98.7 deg.

MISSION EQUIPMENT - Mission equipment (ref: MSFC Payload Description, Level B, October 1973) is as follows:

- Advanced very high resolution radiometer
- Advanced TIROS operational vertical sounder
- Scanning multichannel microwave radiometer electronics
- Scanner
- Microwave radiometer/scatterometer electronics
- Antenna
- Cloud physics radiometer
- Space environmental monitor
- Data collection system.

6.3.3 EOS-F (SEOS)

MISSION REQUIREMENT - The SEOS mission is intended to investigate remote sensing techniques for measuring transient environmental phenomena from a geosynchronous orbit.

MISSION DESCRIPTION - Nominal mission altitude will be 19323 n mi circular at an inclination of 0 deg. Nominal orbit positioning will be 96° W longitude. Nominal mission duration is to be two years with initial launch scheduled for CY 1981. Recovery and/or on-orbit servicing is not planned.

The EOS shall be capable of placing the SEOS experiments in an equatorial orbit of the following characteristics:

- Apogee altitude = 19,323 ± 25 n mi
- Perigee altitude = 19,323 ± 25 n mi
- Inclination = 0 ± 0.2 deg.

The EOS shall place the SEOS experiments at a nominal orbit position of 96° W longitude.

The EOS shall maintain the SEOS experiments on-orbit for not less than two years.

The EOS shall support an initial launch of SEOS experiments in CY 1981.

MISSION EQUIPMENT - Mission equipment is a 1.5-meter aperture telescope.

6.3.4 SEASAT A

MISSION OBJECTIVES - The SEASAT-A mission is designed for development and demonstration of space techniques for forecasting and monitoring sea state, currents, circulation, pileup, storm surges, tsunamis, air/sea interactions, surface winds, and ice formations.

MISSION DESCRIPTION - A nominal orbit altitude of 432 n mi (800 km) is high enough to avoid orbit uncertainties due to drag and low enough to obtain good radar performance with acceptable power consumption. An 108 deg inclination provides good earth coverage, nonsunsychronous, to high latitudes.

MISSION EQUIPMENT - Mission equipment (ref: GSFC Status Presentation, July 1973) is as follows:

- Altimeter radar (topography and wave height)
- Microwave scatterometer (wind speed and direction)

- Microwave radiometer (wind speed, surface temperature, ice images)
- Laser reflectometer
- Synthetic aperture radar (wave spectra and surface images)
- Satellite-to-satellite tracking
- Visible/IR scanner radiometer (surface temperature and features).

6.3.5 SOLAR MAXIMUM MISSION (SMM)

MISSION OBJECTIVES - The basic scientific goal of the SSM is to study the fundamental mechanisms of a solar flare.

MISSION DESCRIPTION - Initial launch is scheduled for June 1978 on a Delta vehicle. Subsequent retrieval and redeployment is planned for Shuttle. Minimum orbital life is one year. The nominal orbit is 275-300 n/mi circular at an inclination of 28-33 deg.

MISSION EQUIPMENT - Mission equipment (ref: GSFS Report X-703-74-72, January 1972) is as follows:

- UV magnetograph
- EUV spectrometer
- High resolution X-ray spectrometer
- Hard X-ray imaging
- Low energy X-ray polarimeter/Medium energy X-ray polarimeter
- Gamma ray detector
- Hard X-ray spectrometer
- Solid state X-ray detector
- Coronograph
- UV spectrometer
- Neutron detector
- H-photometer
- Flare finder.

6.4 FOLLOW-ON MISSION COMPATIBILITY WITH EOS-A

The Report No. 3 Appendix E, Part 2, Trade Study No. 15 presents design cost impacts to the EOS-A spacecraft from the follow-on missions. These impacts are included in Subsection 3.2.17, "Design Growth Economic Study."

The spacecraft components and instruments will be designed and tested to envelope worst-case environmental conditions induced by all anticipated launch vehicles. Therefore, there is no anticipated environmental impact on the spacecraft for follow-on missions.

6.5 OPERATIONS

The EOS prelaunch-readiness tasks, spacecraft/launch vehicle integration, and ascent operations were assessed to establish if launch vehicle selection impacted the cost or complexity of these tasks. The following conclusions have been reached:

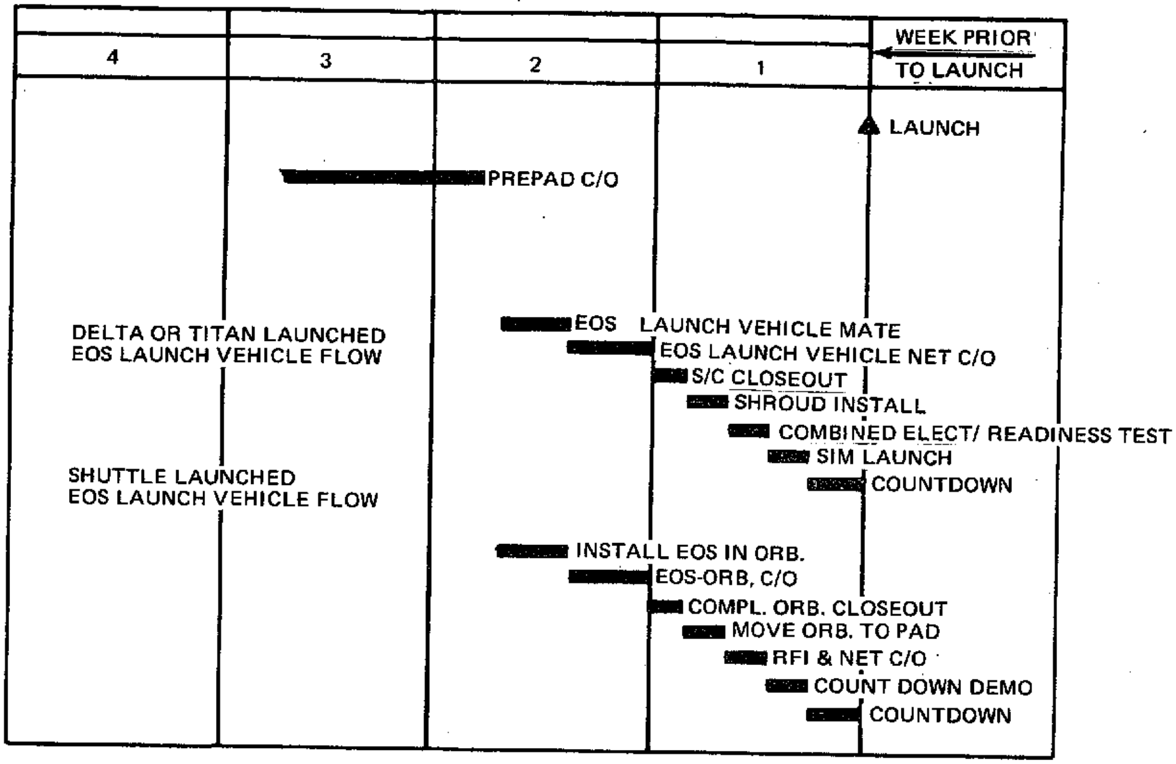
- EOS prelaunch operations are not significantly affected by any of the launch vehicle options studied
- Shuttle ascent and EOS deployment appears more complex than the conventional launch vehicles. However, it does not, when strictly used as a launch vehicle, impact the EOS cost and does reduce EOS risk for ascent operations.

6.5.1 PRELAUNCH OPERATIONS

The prelaunch operations for the EOS at WTR are essentially independent of the launch vehicle selected as far as the basic spacecraft is concerned. The EOS integration into the launch vehicle flow for each of the candidate launch vehicles is shown in Fig. 6-14. In either the Shuttle or conventional launch vehicle, the EOS spends about seven days mated to the launch vehicle prior to launch. The prelaunch check out prior to launch vehicle/EOS mating is the same for the EOS in either case. One consideration worth mentioning is that a Shuttle launch may be of greater schedule risk since it is a Shuttle operational goal to achieve two-week turn around for Shuttle relaunch.

6.5.2 ASCENT

This discussion covers the mission phase from pad liftoff to the placement of EOS into its nominal operational orbit. Since our study utilizes different approaches to place EOS in orbit, three ascents are discussed to cover the range of study options: Delta, Titan, and Shuttle launches.



1-30 **Fig. 6-14 Comparison of EOS - Launch Vehicle Integration and Launch Preparations for Both Conventional Launch Vehicle and Shuttle**

6.5.2.1 DELTA LAUNCH

ASCENT DESCRIPTION - After launch vehicle and EOS spacecraft prelaunch checkout, launch countdown is initiated. EOS mission orbital requirements determine liftoff time, launch azimuth and trajectory to orbit. Launch time is primarily dependent on spacecraft DNTD (9:30 to 12:00). The EOS will be launched by a Delta from WTR to a circular sun-synchronous orbit. The aerodynamic shroud enclosing the EOS spacecraft will be ejected after passing through the region of significant aerodynamic loads. Launch tracking is provided by WTR for 6 to 8 min which covers the termination of the second-stage first burn. The space vehicle will be on a transfer orbit and coast to apogee where the launch vehicle second stage will be ignited for a second burn, inserting the space vehicle into EOS mission orbit. During the coast period while waiting to pass over the Fairbanks tracking site, the launch vehicle will hold a favorable attitude for EOS separation. Upon acquisition of EOS by Fairbanks the ground will prepare to backup the automatic separation command. After the EOS is separated the ACS logic and attitude control will be enabled and the Delta launch vehicle will perform an evasive maneuver.

DESIGN CONSIDERATIONS - Spacecraft design considerations are:

- Structure - During launch the spacecraft will experience its highest load condition, therefore Delta loads will dictate the structural design margins
- Electrical Power - The spacecraft will have to supply its own power after launch umbilical release, therefore batteries must be sized to provide power until the solar panels are deployed in the EOS mission orbit
- Command and Attitude Control - Upon release from the launch vehicle the EOS must provide attitude control and be capable of being commanded. This requires that the communication receiver must be on and that the attitude control functions are enabled.

6.5.2.2 TITAN LAUNCH

ASCENT DESCRIPTION - Titan operations will be similar to the Delta up to the time that the space vehicle is inserted into transfer orbit. After the transfer orbit operations are completed preparations will be made to separate EOS from the launch vehicle. The Titan will go to a favorable attitude for EOS separation. Upon acquisition of EOS by the tracking net the ground will prepare to backup the automatic separation commands. After the EOS is separated the ACS logic and attitude control will be enabled and the Titan launch vehicle will perform an evasive maneuver. The EOS now coasts to apogee where it will utilize a single SRM for mission orbit insertion. The insertion burn will be performed during ground station coverage to provide backup for the onboard automated sequences.

DESIGN CONSIDERATIONS - The same Delta considerations apply to the Titan launch vehicle.

SRM - Since the SRM thrust and burn time is fixed, propellant loading will be tailored to obtain the required ΔV. The thermal design must consider heat soak back from the expended SRM.

6.5.2.3 SHUTTLE LAUNCH

ASCENT DESCRIPTION - The Shuttle and EOS will be launched from WTR to achieve mission orbit. Solid Rocket Booster staging will occur approximately 2 min, 7 sec into ascent. MECO will occur approximately 8 min, 9 sec into ascent and at a lower velocity than for an ETR launch. This condition is dictated by the proximity of the External Tank impact point into the Pacific Ocean. External Tank separation is suborbital and occurs at approximately 8 min, 32 sec. The OMS will provide additional ΔV required to insert the orbiter and EOS into mission altitude. OMS kits will be added as required to achieve orbit insertion and to fulfill on-orbit and de-orbit maneuvers. The first 6 to 8 min of tracking will be provided by WTR. After post-insertion Orbiter checkout, the EOS will be checked out and deployed. The Orbiter will then maneuver to a station keeping distance and verify EOS operations.

DESIGN CONSIDERATIONS - Spacecraft design considerations are:

- Structure - Cradle support in orbiter payload bay; remote manipulation, grab-type and effector installation; deployment/retrieval/docking interfaces
- Electrical - Power, signal and data interfaces for activation/deactivation and check out
- Operations - RF control of EOS while operating in the vicinity of orbiter during deployment release and retrieval.

6.6 RELIABILITY-HISTORY OF SUCCESSES AND FAILURES

Reliability evaluations of the Delta 2910, and Titan IIIB launch vehicles were performed. The 50 and 90% confidence level success probabilities for each vehicle are as follows:

			Reliability		
Booster	Launches	Failures	Best Est. 50%C.L.	90% C.L.	Comments
Delta 2910	101	10	0.89	0.85	1st stage failures - 2 2nd stage failures - 4 3rd stage failures - 4
Titan III B	45	0	0.96	0.91	39 successful launches without failure since 1967

6.7 LAUNCH VEHICLE ENVIRONMENTS

The spacecraft will be exposed to flight dynamic environments from liftoff to separation of the spacecraft from the launch vehicle. These environments, predominantly launch vehicle dependent, are presented for the following four boosters:

- Delta 2910
- Delta 3910
- Titan IIIB/NUS
- Shuttle.

6.7.1 LIMIT LOAD FACTORS

The spacecraft will experience steady state accelerations due to engine thrust and, in addition, dynamic accelerations due to engine ignition, shutdowns, and POGO. The maximum (steady state and dynamic) limit load factors for Delta, Weight Constrained Titan III, Titan IIIB, and Shuttle Launch Vehicles are shown in Table 6-10, 6-11, and 6-12, respectively. Figure 6-15 shows the spacecraft coordinate system sign convention.

Table 6-10 Limit Load Factors – Delta 2910 and Delta 3910

CONDITION	LONGITUDINAL X	LATERAL Y OR Z
LIFT-OFF	+ 2.9 - 1.0	2.0
MAIN ENGINE CUTOFF	+ 12.3	0.65

T1-9

Table 6-11 Limit Load Factor – WTR Titan III B/NUS Launch Vehicle

CONDITION	LONGITUDINAL X	LATERAL Y OR Z
LIFT-OFF	+ 2.3 - 0.8	2.0
STAGE I SHUTDOWN (DEPLETION)	+ 8.2 - 2.5	1.5
STAGE II SHUTDOWN (COMMAND)	+ 10.8 - 2.0	1.5

NOTES:
1. LOAD FACTOR CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD.
2. INCLUDES BOTH STEADY STATE AND DYNAMIC CONDITIONS.

T1-10

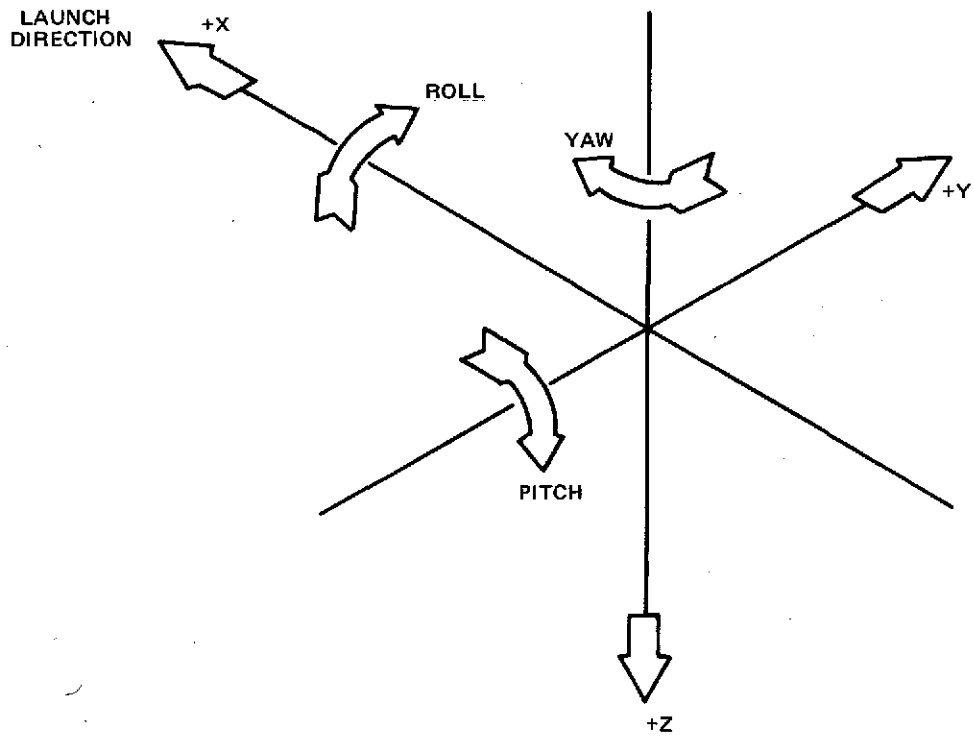
Table 6-12 Limit Load Factors – Payload Bay – Shuttle

CONDITION	DIRECTIONS (3)		
	X	Y	Z
LIFT-OFF (1)	+ 1.7 ± 0.6	± 0.3	+ 0.8 + 0.2
HIGH Q BOOST	+ 1.9	± 0.2	- 0.2 + 0.5
BOOSTER END BURN	+ 3.0 ± 0.3	± 0.2	+ 0.4
ORBITER END BURN	+ 3.0 ± 0.3	± 0.2	+ 0.5
SPACE OPERATIONS	+ 0.2 - 0.1	± 0.1	± 0.1
ENTRY	± 0.25	± 0.5	- 3.0 + 1.0
SUBSONIC MANEUVERING	± 0.25	± 0.5	- 2.5 + 1.0
LANDING AND BRAKING	± 1.5	± 1.5	- 2.5
CRASH (ULTIMATE) (2)	- 9.5 + 1.5	± 1.5	- 4.5 + 2.0

NOTES
1. THESE FACTORS INCLUDE DYNAMIC TRANSIENT LOAD FACTORS.
2. THESE FACTORS ARE ULTIMATE AND ONLY USED TO DESIGN PAYLOAD SUPPORT FITTINGS. THE SPECIFIED CRASH LOAD FACTORS SHALL ACT SEPARATELY.
3. LOAD FACTOR CARRIES THE SIGN OF THE EXTERNALLY APPLIED LOAD. POSITIVE X, Y, Z DIRECTIONS EQUAL FORWARD, RIGHT AND DOWN.

T1-11

1



1-31 **Fig. 6-15 Spacecraft Coordinate System**

6.7.2 ACOUSTIC FIELD

The critical flight periods for this environment are at launch and transonic flight regimes. The maximum expected composites of the launch and transonic flight acoustic levels for each of four launch vehicles are shown in Table 6-13.

Figure 6-16, a comparison of the acoustic spectra (octave band sound pressure levels), indicate that the Shuttle acoustic field is the highest.

Table 6-13 Maximum Expected Flight Acoustic Level (Internal)

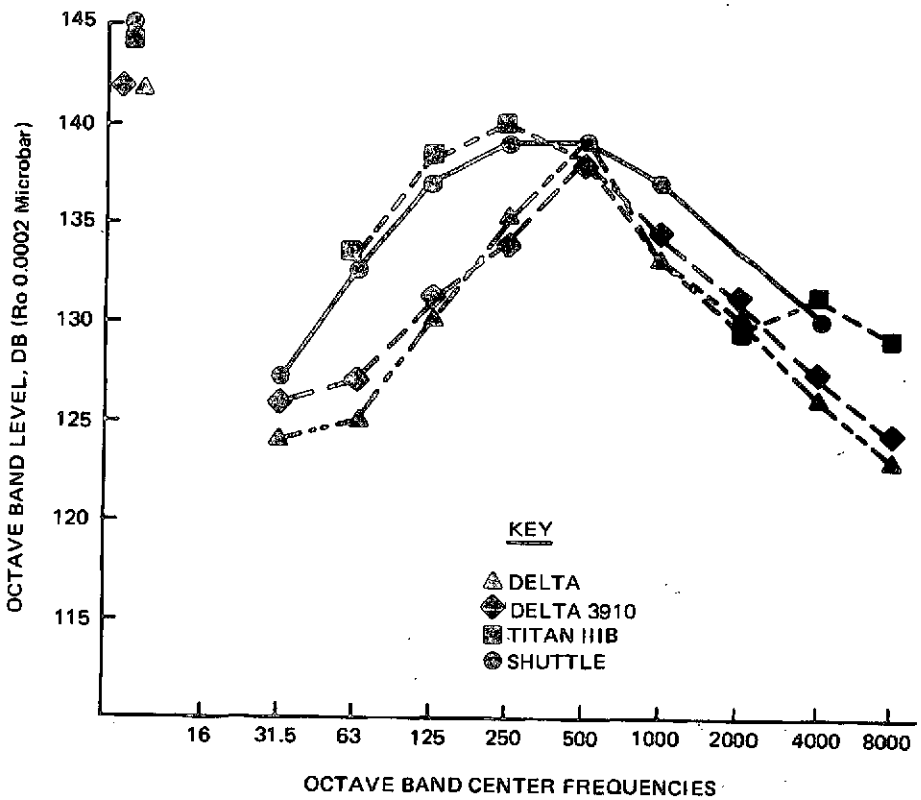
CENTER FREQUENCY, Hz	OCTAVE BAND SOUND PRESSURE LEVELS (dB RE: 20 μ NEWTON/m²)			
	DELTA 2910[1]	DELTA 3910 [1]	TITAN IIIB[2]	SHUTTLE[3]
31.5	124	126	–	127
63	125	127	133	132.5
125	130	131	138	137
250	135	134	140	139
500	139	138	138	139
1000	133	134	133	137
2000	130	131	129.5	133.5
4000	126	127	131	130
8000	123	124	129	–
OVERALL	142	142	144.5	145
DURATION, SEC	45	45	60	30

T1-12

NOTES:

(1) DELTA 2.44m (96 IN.) DIAMETER FAIRING WITH ACOUSTIC INSULATION
(2) LMSC 3.05 m (120 IN.) DIAMETER FAIRING
(3) WITHOUT ACOUSTIC ATTENTUATION TREATMENT OF PAYLOAD BAY AREA

1



1-32 Fig. 6-16 Maximum Expected Flight Acoustic Levels (Internal)

DELTA 2910 AND DELTA 3910 - The acoustic levels shown are external to the MDAC 2.44m (96 inch) diameter fairing with an acoustic insulating blanket.

TITAN IIIB - The acoustic levels shown are internal to the LMSC P-123, 3.05m (120 in.) diameter fairing.

SHUTTLE - The acoustic levels shown are internal to the Shuttle paylaod bay without any potential acoustic attenuation treatment.

6.7.3 RANDOM VIBRATION

The acoustic field at launch and transonic flight regimes generate random vibrations of the launch vehicle airframe, and spacecraft fairing. A portion of this random vibration is structure-borne transmitted to the spacecraft. The maximum expected flight structure-borne random vibration envelopes, at the spacecraft/launch vehicle interface for each of four launch vehicles are shown in Table 6-14, 6-15, 6-16 and 6-17 for Delta 2910, Delta 3910, Titan IIIB, and Shuttle respectively. The Shuttle spectrum is included for completeness only but is not used in the comparison because it represents levels for unloaded structure (i.e., does not consider the spacecraft mass attenuation effect).

Table 6-14 Maximum Expected Flight Random Vibration — Delta 2910 Launch Vehicle

FREQUENCY RANGE, Hz	ACCELERATION SPECTRAL DENSITY, g^2/Hz	ACCELERATION OVERALL, g-rms	DURATION PER AXIS, SEC
20-300 300-1000 1000-2000	+ 3dB/OCT 0.03 - 3dB/OCT	6.8	45
NOTES: (1) INPUT AT BASE OF SPACECRAFT ATTACH FITTINGS (2) WITH FAIRING ACOUSTIC INSULATION			

T1-13

Table 6-15 Maximum Expected Flight Random Vibration — Delta 3910 Launch Vehicle

LEVEL	FREQUENCY RANGE, Hz	ACCELERATION SPECTRAL DENSITY, g^2/Hz	ACCELERATION OVERALL, g-rms	DURATION PER AXIS, SEC
"A"	20-300 300-1000 1000-2000	+ 3dB/OCT 0.04 - 3dB/OCT	7.9	10
"B"	20-300 300-1000 1000-2000	+ 3dB/OCT 0.03 - 3dB/OCT	6.8	35
NOTES: (1) INPUT AT BASE OF SPACECRAFT ATTACH FITTINGS (2) WITH FAIRING ACOUSTIC INSULATION				

T1-14

Table 6-16 Maximum Expected Flight Random Vibration — Titan III B Launch Vehicle

SPACECRAFT WEIGHT, LB	FREQUENCY RANGE, Hz	ACCELERATION SPECTRAL DENSITY, g^2/Hz	ACCELERATION OVERALL, g-rms	DURATION PER AXIS, SEC
3000	20-500 500-1000 1000-2000	+ 3dB/OCT 0.07 - 6dB/OCT	9.4	60
⩾6750	20-50 500-1000 1000-2000	+ 3dB/OCT 0.02 - 6dB/OCT	5.0	60
NOTES: (1) INPUT AT BASE OF SPACECRAFT ATTACH FITTINGS				

T1-15

Table 6-17 Maximum Expected Flight Random Vibration — Shuttle

LEVEL	FREQUENCY RANGE, Hz	ACCELERATION SPECTRAL DENSITY, g^2/Hz	ACCELERATION OVERALL, g-rms	DURATION PER AXIS, SEC
"A" LIFT-OFF	20-90 90-300 300-2000	+ 6dB/OCT 0.10 - 6dB/OCT	7.0	21
"B" MAX Q-TRANSONIC	20-40 40-150 150-2000	+ 6dB/OCT 0.05 - 6dB/OCT	3.6	9
NOTES: (1) AT MID-FUSELAGE MAIN LONGERON (2) UNLOADED LEVELS — PAYLOAD MASS ATTENUATION EFFECT NOT CONSIDERED				

T1-16

Figure 6-17, comparing the random vibration spectra, indicates that the Titan IIIB levels are the highest.

6.7.4 SINUSOIDAL VIBRATION

The sinusoidal vibration environment is an envelope of launch vehicle responses, at the spacecraft/launch vehicle interface, resulting from excitation of the launch vehicle low frequency modes due to various forcing functions (i.e., POGO, engine ignition, engine shutdown and sinusoidal transients occurring throughout the flight). The maximum expected flight sinusoidal vibration envelopes of each of four launch vehicles are shown in Table 6-18.

Table 6-18 Maximum Expected Flight Sinusoidal Vibration

	DELTA 2910 & DELTA 3910		TITAN IIIB		SHUTTLE	
	FREQUENCY RANGE, Hz	ACCELERATION ZERO-TO-PEAK, g	FREQUENCY RANGE, Hz	ACCELERATION ZERO-TO-PEAK, g	FREQUENCY RANGE, Hz	ACCELERATION ZERO-TO-PEAK, g
LONGITUDINAL	5 - 9.5 9.5 - 15 15 - 21 21 - 200	8.4 mm d.a. ± 1.5 ± 4.0 ± 1.5	5 - 20 20 - 50 50 - 200	15.2 cm/sec ± 2.0 ± 1.5	5 - 35	± 0.25
LATERAL	5 - 7.1 7.1 - 14 14 - 200	12.7 mm d.a. ± 1.3 ± 1.0	5 - 13 13 - 22 22 - 200	3.8 mm d.a. ± 1.3 ± 1.0	5 - 35	± 0.25
SWEEP RATE, OCT/MIN	4.0		4.0		1.0	

T1-17

NOTE: INPUT AT BASE OF SPACECRAFT ATTACH FITTINGS

Figures 6-18 and 6-19 compare the longitudinal and lateral sinusoidal vibration levels, respectively. Shuttle levels are least significant. The Delta levels are highest in the frequency ranges 6 to 21 Hz and 5 to 12 Hz for longitudinal and lateral axes, respectively. The Titan levels exceed the Delta by 0.5 g at 21 to 50 Hz and by 0.3 g at 14 to 22 Hz for longitudinal and lateral axes, respectively.

NOTE

In the absence of Titan IIIB sinusoidal vibration levels (not specified by Martin-Marietta), the Titan IIIC levels specified by GSFC (S-320-G01) are used.

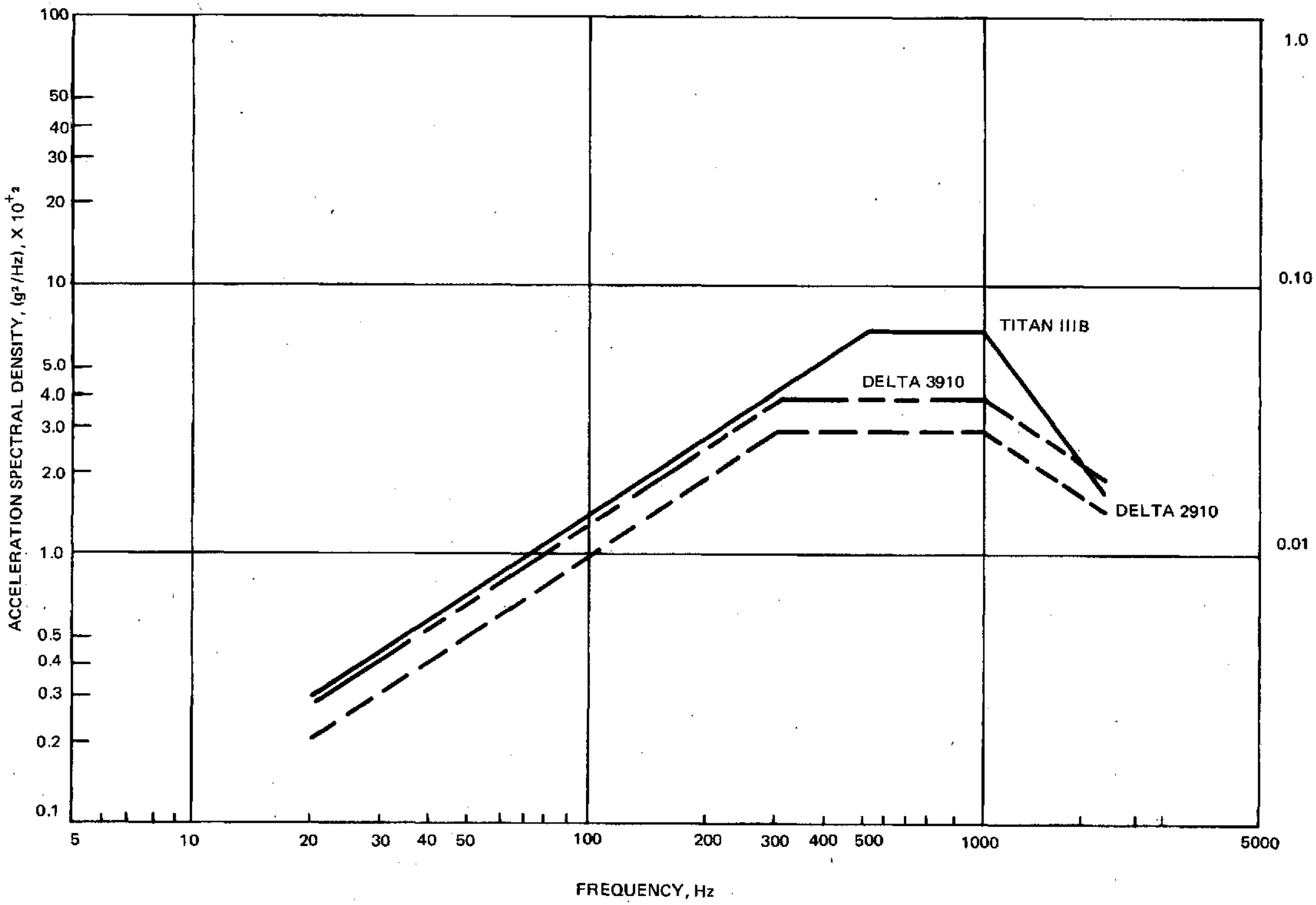


Fig. 6-17 Random Vibration

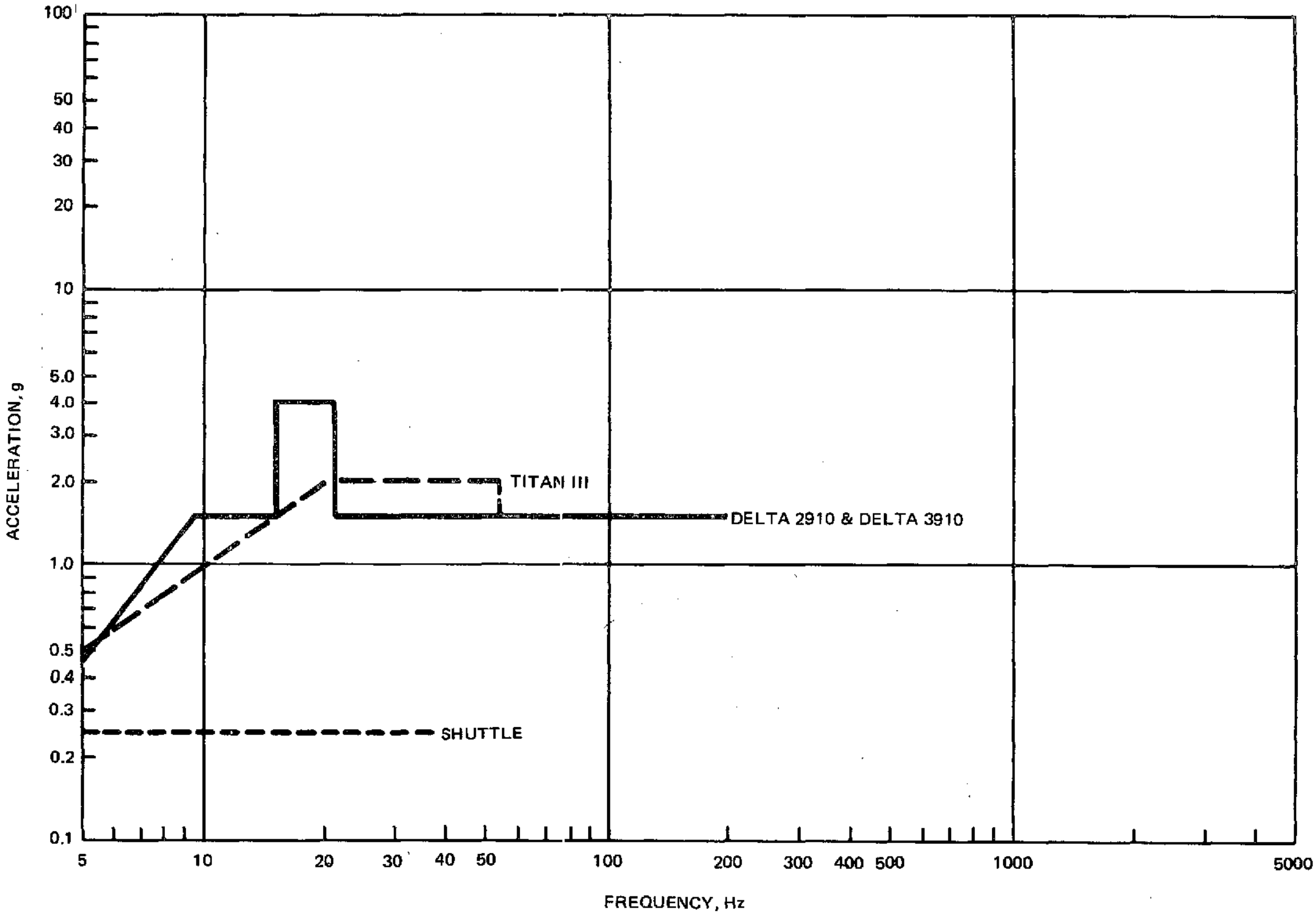
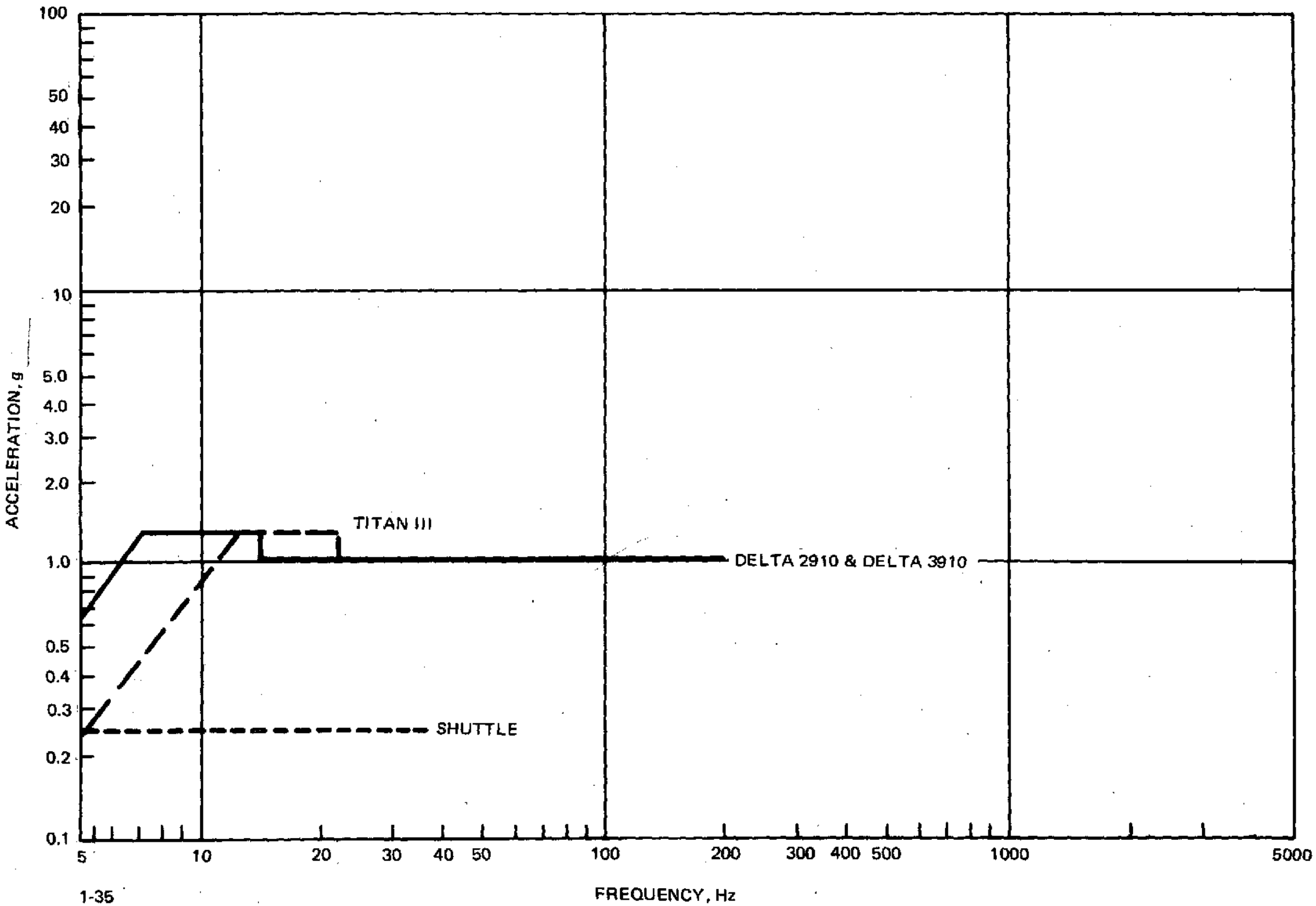


Fig. 6-18 Longitudinal Sinusoidal Vibration



1-35

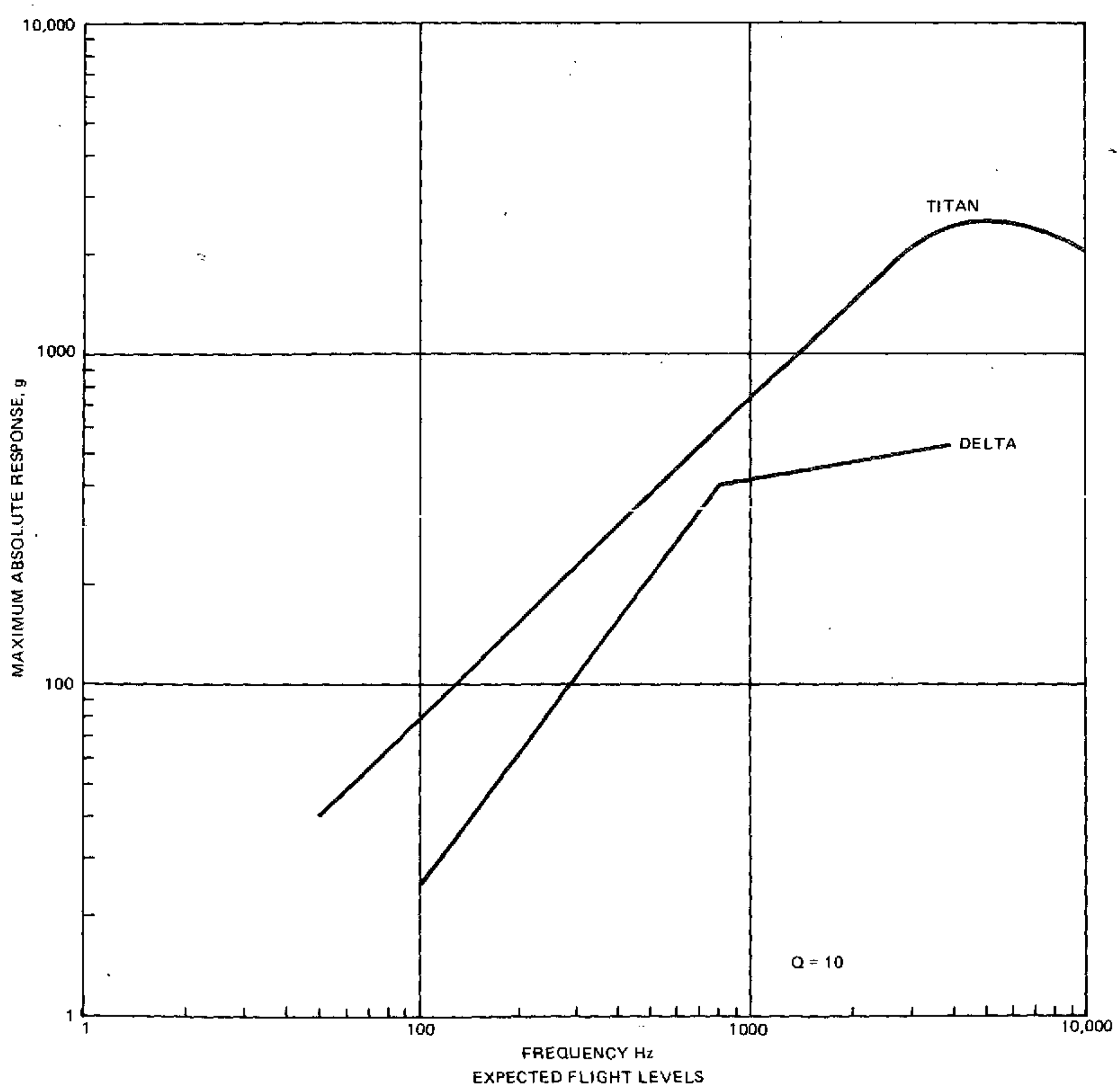
Fig. 6-19 Lateral Sinusoidal Vibration

1

6.7.5 SHOCK

Shock impulses are transmitted to the spacecraft at separation of the launch vehicle stages, at engine ignition, at separation of the fairing, and at separation of the spacecraft from the launch vehicle.

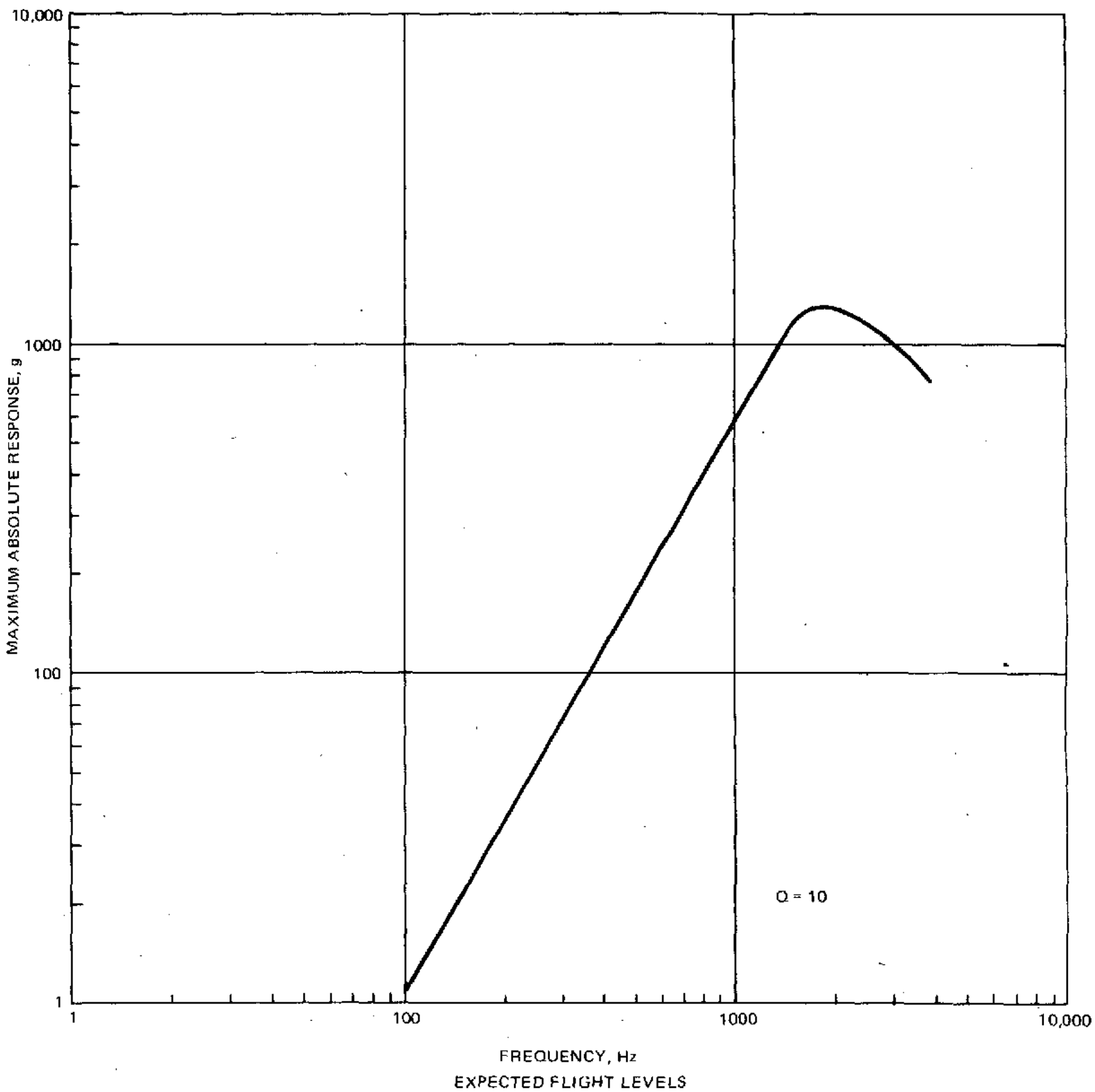
Launch Vehicle Induced Shocks - The maximum launch vehicle induced shock experienced by the spacecraft is defined by the shock response spectra, at the spacecraft/launch vehicle interface, shown in Fig. 6-20 for Delta and Titan. Shuttle induced shock are not specified.



1-36 Fig. 6-20 Shock Response Spectrum at Spacecraft/Launch Vehicle Interface – Launch Vehicle Induced Shocks

Spacecraft Separation Shock - This shock event is independent of the launch vehicle. Separation of the spacecraft from the launch vehicle is usually initiated by pyrotechnic devices which disengage a clamp (i.e., Marman clamp). A typical shock response spectrum, at the spacecraft/launch vehicle interface is shown in Fig. 6-21.

Shuttle Landing Shock - This shock event is represented by the shock pulses shown in Table 6-19. Consideration should be given to analyzing the landing shock environment in lieu of imposing a test requirement.



1-37

Fig. 6-21 Separation Shock Response Spectrum at Spacecraft/Launch Vehicle Interface – Typical Marman Clamp Separation System

Table 6-19 Shuttle Landing Shock* Rectangular Pulses

NUMBER OF APPLICATIONS	PULSE DURATION, ms	ACCELERATION, g
22	170	.23
37	280	.28
22	330	.35
20	360	.43
9	350	.56
4	320	.72
1	260	1.50
*VERTICAL UP DIRECTION (-Z)		

TI-18

6.7.6 MINIMUM FREQUENCY REQUIREMENTS

To avoid dynamic coupling between the low-frequency launch vehicle and spacecraft modes, the minimum frequency criteria for the spacecraft constrained at the spacecraft/launch vehicle interface is specified in Table 6-20. The Delta minimum frequency criteria are the highest.

Table 6-20 Minimum Frequency Criteria

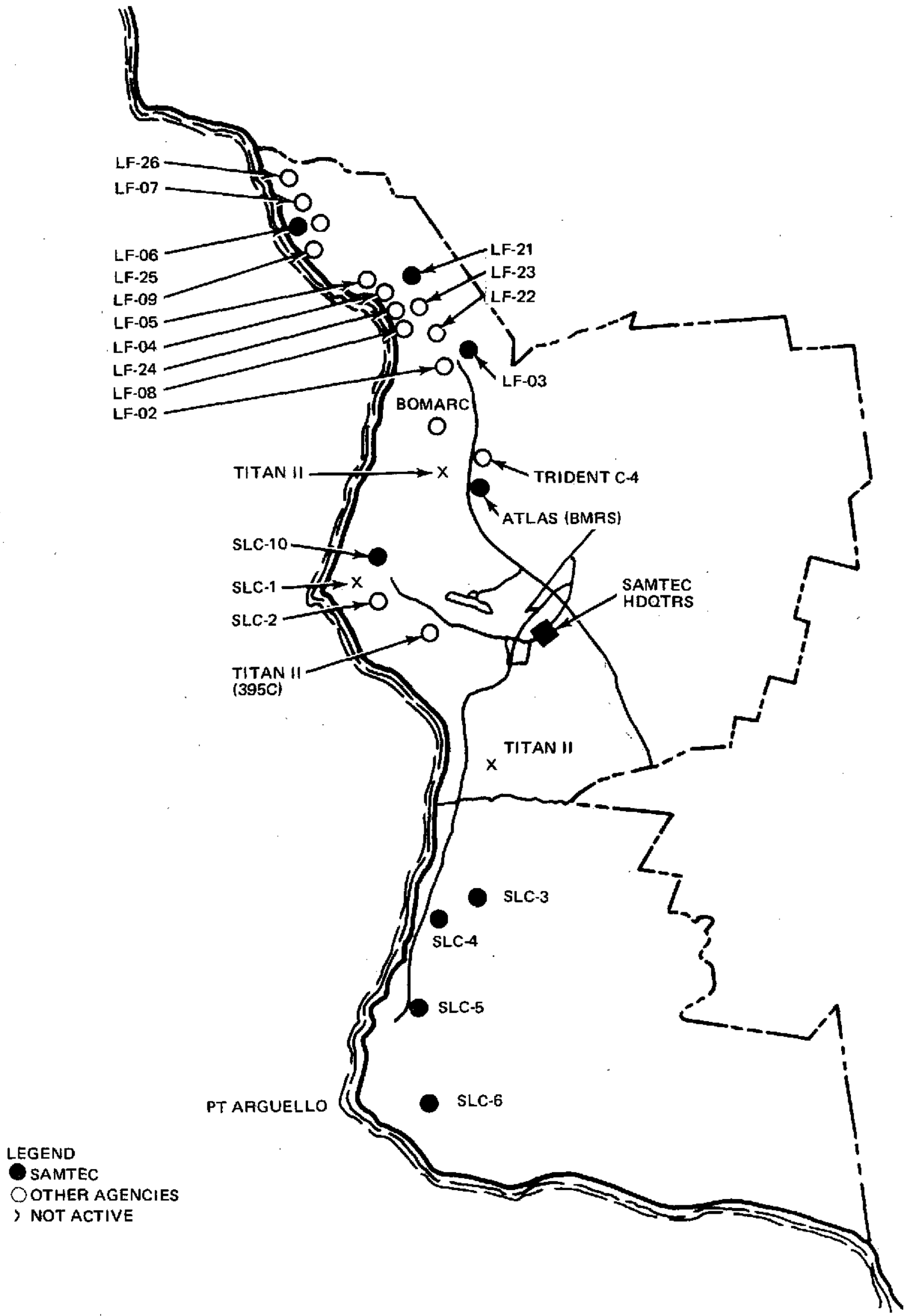
LAUNCH VEHICLE	MINIMUM FREQUENCY, Hz	
	LONGITUDINAL	LATERAL
DELTA 2910	35	15
DELTA 3910	35	15
TITAN IIIB	20	10
SHUTTLE	N.D.	N.D.
N.D. NOT DEFINED.		

T1-19

6.8 LAUNCH SITE SAFETY CONSIDERATIONS

Any vehicles launched from WTR must satisfy the safety requirements of SAMTECM 127-1, Range Safety. Facility constraints, launch azimuth limitations, destruct system requirements, trajectory characteristics (dispersions, velocity vector turn capability, land overflight and typical failure modes) and ground safety requirements must be met.

Figure 6-22 identifies SLC 2W as the Delta 2910/3910 launch site and SLC 4 as the Titan launch site. The proposed launch azimuth is 204°W. Since each boost stage has a dedicated facility, it must be assumed that compatability requirements will be satisfied. Launch vehicles destruct systems are standard booster equipment.



1-38

Fig. 6-22 Launch Facilities -- Vandenberg Air Force Base

Launch trajectory characteristics must be determined during the next study phase. The extensive experience of launch vehicle contractors in solving ground safety problems, especially ordinance and hypergolics, places ground safety requirements in the "easily accomplished" category.

The Shuttle WTR launch site is ill-defined at present. However, the requirements to make EOS compatible with a manned vehicle enhances range safety. Vehicle command and control and constant monitoring during the time EOS is in the cargo bay are the prime reasons. Shuttle abort capability also is a consideration.

The EOS has hazard sources that require dedicated study, but are also simply resolved. Pressurized tankage is used for the ACS; either GN_2 at approximately 3000 psi or hydrazine at approximately 400 psi. With appropriate tank safety factors equal to or greater than two, and leakage detection and safing capability, rupture, leakage, and spills are adequately controlled. Where SRM's are required, standard electromechanical safe/arm devices assure against any ordnance hazards. Lastly, by loading EOS ACS propellants prior to stacking the payload at the launch site, system integrity is established early and limits any leakage risks to a facility designed to handle it.

6.9 REFERENCES

6.9-1 JSC International Note No. 74-FM-5, "Effects Of An Elliptic Servicing Orbit On Orbiter Rendezvous With The Goddard Earth Observation Satellite," dated 29 January 1974.

6.9-2 JSC Internal Note No. 74-FM-6, "EOS Maneuvering To A Shuttle Compatible Servicing Orbit Prior to Shuttle Lift-Off," dated 4 February 1974.

6.9-3 JSC Internal Note No. 74-FM-17, "Preliminary Representative Mission Profile And Performance Analysis For A Typical EOS Servicing Mission," dated 7 March 1974.

RECOMMENDATIONS

7 - RECOMMENDATIONS

With the advent of Shuttle for deployment, retrieval, servicing or resupply of space missions, Shuttle utilization becomes an important driver in the design and operation of EOS. Shuttle/EOS missions become feasible for EOS mission orbit altitudes below about 400 n mi (740 km). To obtain full tracking from Sioux Falls over CONUS, and also since orbit decay due to aerodynamic drag becomes excessive below about 350 n mi (650 km), it is recommended that EOS be orbited at an altitude within the range 365 to 385 n mi. The altitude within this range should be optimized for the selected sensor instrument swath width.

It is further recommended that the earlier EOS-A spacecraft be designed to be launch compatible with the Delta 2910 launch vehicle. The EOS preliminary design weights shown in this report indicate that this is feasible.