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CONTINGENCY STUDY FOR THE THIRD INTERNATIONAL SUN-EARTH EXPLORER (ISEE-3) SATELLITE

Prepared For

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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By

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Under

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Prepared by:

David W. Dunham 5/4/79
D. W. Dunham Date
Task Leader

Approved by:

Richard W. Herder 5/4/79
R. W. Herder Date
Section Manager

R. D. Headrick 5-?-?
R. D. Headrick Date
Department Manager

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ABSTRACT

The third satellite of the International Sun-Earth Explorer program has been inserted into a periodic halo orbit about L_1, the collinear libration point between the Sun and the Earth-Moon barycenter. This document presents a plan that was developed to enable insertion into the halo orbit in case there had been a large underperformance of the Delta second or third stage during the maneuver to insert the spacecraft into the transfer trajectory. After one orbit of the Earth, a maneuver would be performed near perigee to increase the energy of the orbit. A relatively small second maneuver would put the spacecraft in a transfer trajectory to the halo orbit, into which it could be inserted for a total ΔV cost within the fuel budget. Overburns ("hot" transfer trajectory insertions) were also studied.

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SECTION 1 - INTRODUCTION

The International Sun-Earth Explorer Mission is a three-spacecraft joint NASA - European Space Agency (ESA) program to monitor the Earth's magnetosphere and the interplanetary medium (References 1 and 2). The third spacecraft, ISEE-3, was planned to measure the solar wind from a halo orbit about the L_1 libration point between the Earth and the Sun. Strategies have been designed to maximize the scientific return of ISEE-3 for several cases of possible poor engine performance during injection into the transfer trajectory. Many of the basic ideas behind these strategies can be used for contingency planning for other missions to the vicinity of the collinear libration points of the Sun-Earth system.

The Goddard Mission Analysis System (GMAS) (Reference 3) was used to study contigency situations which might occur. It was initially believed that it would be necessary to use a gravitational assist from the Moon, probably involving a close lunar swingby, to correct a large underperformance ("cold" burn) at transfer trajectory insertion (see Reference 4). Software was developed for targeting to the halo orbit using a lunar gravity assist (Reference 5 and 6). However, a three-impulse strategy was developed which can correct a wide variety of contigency situations without the need for a lunar swingby. The spacecraft is allowed to complete one orbit. The first maneuver is performed at the first perigee to raise apogee to libration-point distances (about 1.5×10^6 km). Shadow constraints and penalty factors due to burn duration must be computed for the perigee maneuver. The second maneuver is performed several days later to define a transfer trajectory to the halo orbit, into which the spacecraft is inserted with the third maneuver. Details of the procedure, including problems which sometimes arise due to lunar perturbations, are given in Section 2. The results of a study of large overburns at transfer trajectory insertion are described in Section 3. The overall strategy is summarized in Section 4.

SECTION 2 - STRATEGY TO CORRECT LARGE UNDERBURNS

A three-impulse strategy which can be used to correct most possible large underburns at transfer trajectory insertion (TTI) has been outlined in the introduction. The one-sigma (σ) standard deviation for the velocity impulse imparted by the Delta rocket at the ISEE-3 TTI has been calculated to be 5.6 meters/second, the value used for this investigation. A contingency situation was provisionally defined to exist when the velocity at TTI was -3σ or less than the nominal planned TTI velocity (V). An underburn significantly decreases the energy of the transfer trajectory, causing the spacecraft to go into a highly elliptical orbit about the Earth without reaching the vicinity of the L_1 libration point. Since the expected pointing errors (ϵ_p) are small, they have been virtually ignored in this study. They contribute to the error of the energy of the transfer trajectory only to the second order, being equivalent to a velocity error of $V (1 - \cos \epsilon_p)$. Our studies have shown that reasonably large pointing errors can be corrected during a midcourse correction (MCC) maneuver about a day after TTI.

A comparison of the three-impulse strategy with direct transfers is described in Section 2.1. Maneuvers which may be required during the first orbit are discussed in Section 2.2. Details about the crucial burn at first perigee are given in Section 2.3. The new transfer trajectory to the halo orbit, including the necessary midcourse correction ("second" impulse), is discussed in Section 2.4. Results of the GMAS contingency studies for ISEE-3 for three launch dates are tabulated in Section 2.4. Postponed transfers are described in Section 2.5.

2.1 COMPARISON OF THE THREE-IMPULSE STRATEGY WITH DIRECT TRANSFERS

An optimized three-impulse trajectory for a -3σ velocity error is plotted in Figure 2-1, using the Rotating Libration Point (RLP) coordinate system, which

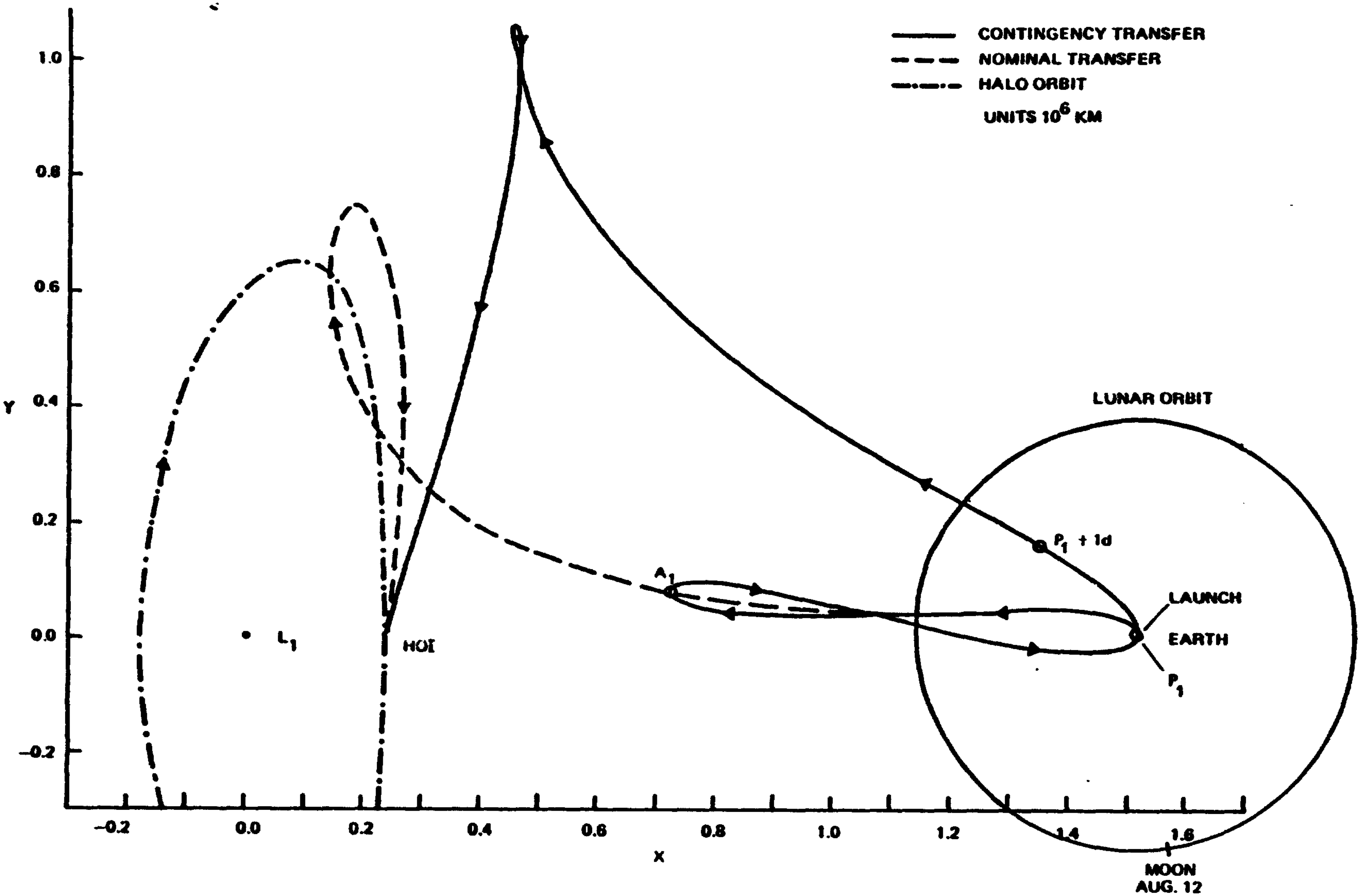

Figure 2-1. V-3σ Contengency Case, Launch August 12, 1978

is centered at L_1 and rotates at the solar angular rate. The X-Y plane is the ecliptic plane. The Earth-Moon barycenter is on the positive X-axis, but the distance varies slightly during the year due to the eccentricity of the orbit of the Earth-Moon barycenter. The first orbit has a period of 30.1 days and is twisted into a figure-8 shape by the rotation of the coordinate system. If no maneuvers were performed, the spacecraft would describe further figure-8 orbits, the axis of each rotated about 30 degrees from the axis of the previous one due to the rotation of the coordinate system. (This effect is presented in Figure 2-2.) However, the maneuver at first perigee (P_1) boosts the apogee to a distance comparable to L_1 and another maneuver a day later defines a new trajectory to the halo orbit, which has a Z-amplitude of 110,000 km and is shown by the stippled curve in Figure 2-1. For comparison, the planned nominal transfer trajectory, with no error at TTI, is shown as a dashed line. Due to the rotation during the 30.1 days in the Earth orbit, the new transfer trajectory goes much further above the X-axis than the nominal transfer trajectory and enters the halo orbit at a steeper angle, raising the cost of the halo orbit insertion (HOI) maneuver. The total ΔV cost for the three maneuvers is 216 meters/second, which is a large portion of the ISEE-3 fuel budget, but not as high as expected.

Paradoxically, the situation improves with larger errors. The trajectory for the optimized $V - 6\sigma$ case is plotted in Figure 2-3. The first orbit now cusps at apogee rather than describing a figure-8, and its period is 18.3 days. A slightly larger maneuver is needed at P_1, but this is more than compensated for by the reduction in costs for the other two maneuvers caused by the smaller rotation during the first orbit. The total ΔV cost is only 141 meters/second. Except for some cases strongly perturbed by the Moon, as discussed in Section 2.4.3, the total impulsive ΔV costs remain below 180 meters/second from $V - 6\sigma$ to $V - 20\sigma$. For velocity errors algebraically less than about -20σ, the first perigee maneuver costs become prohibitive. For such cases, the apogee is under 250,000 km, far short of the Moon's orbit.

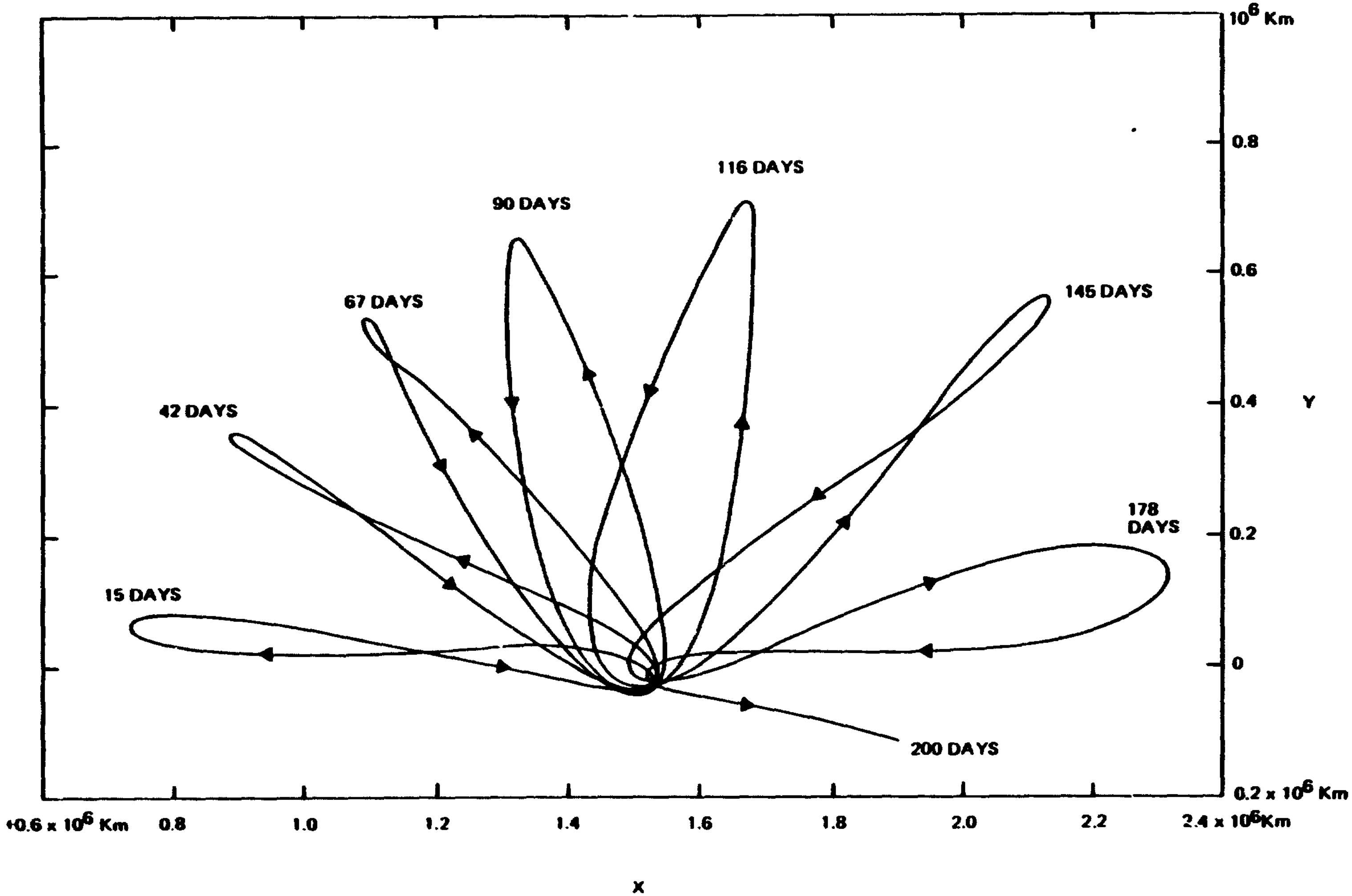

Figure 2-2. Uncorrected V-3σ Trajectory for July 23, 1978 Launch (RLP Coordinates)

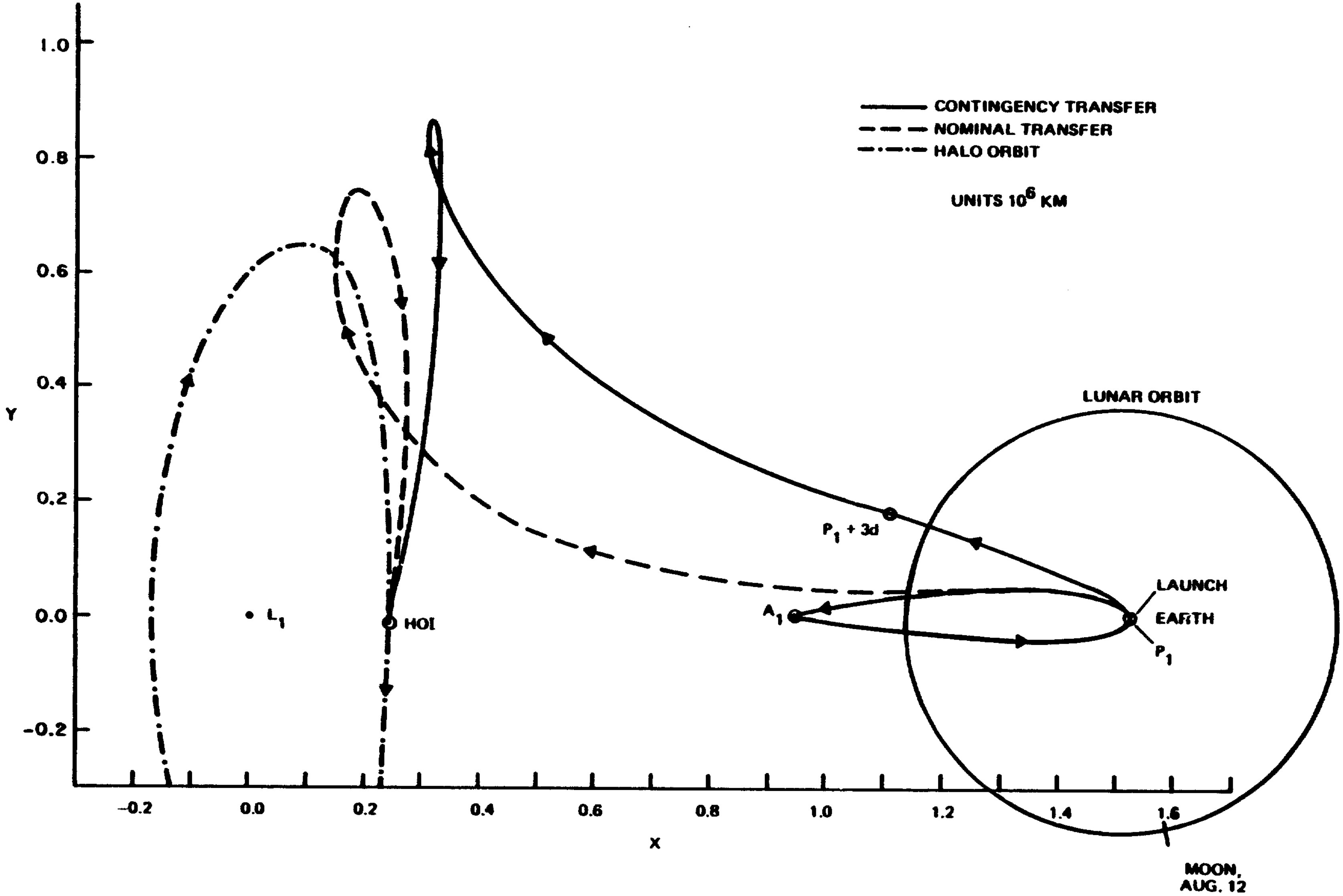

Figure 2-3. V-6σ Contingency Case, Launch August 12, 1978

The costs for correcting velocity errors for direct transfers to the halo orbit are listed in Table 2-1. A large midcourse correction is applied 18^h after TTI. The halo orbit insertion costs decrease slightly with greater errors. Ecliptic plane (ΔV_{XY}) and ecliptic normal ($\dot{Z}$) components of the impulsive ΔVs are given in the table, where ΔV_{2XY} is the in-plane component of the mid-course correction, and $\dot{Z}$ is the out-of-plane component. The total ΔV for the MCC is given by ΔV_2, where

$$\Delta V_2 = \sqrt{\Delta V_{2XY}^2 + \dot{Z}_2^2}$$

Also, ΔV_{IXY} is the in-plane component of the HOI ΔV, $\dot{Z}_{IN}$ is the out-of-plane component, and the total HOI ΔV is ΣV_{IN}, where

$$\Sigma V_{IN} = \Delta V_{IXY} + |\dot{Z}_{IN}|$$

and the total ΔV costs, $T\Sigma V$, which were minimized is given by $T\Sigma V = \Delta V_2 + \Sigma V_{IN}$. For the MCC, it was assumed that the spacecraft would be tilted to allow the full ΔV be applied with the radial jets, so the vector sum was used. For the HOI, the spacecraft axis is kept near the ecliptic normal, so that ΔV_{IXY} is applied by the radial jets, and $\dot{Z}_{IN}$ is applied by the axial jets. The configuration of the radial and axial hydrazine jets of ISEE-3 is described in Reference 9.

As expected, the total costs for correcting TTI errors for the direct transfers increase in proportion to the size of the error. It is well known that an increase in the semimajor axis (Δa) of an elliptical orbit is proportional to the

Table 2-1. Direct Transfer Costs for V-nσ for August 12, 1978 Launch

n	MCC AT	MCC			HOI			TΣV (m/sec)
		ΔV_{2XY} (m/sec)	$\dot{Z}_2$ (m/sec)	ΔV_2 (m/sec)	ΔV_{IXY} (m/sec)	$\dot{Z}_{IN}$ (m/sec)	ΣV_{IN} (m/sec)	
3	18^h	92.4	11.0	93.1	32.4	2.8	35.2	128.3
4	18^h	123.8	14.8	124.7	30.5	−3.8	34.3	159.0
5	18^h	155.5	18.7	156.6	28.6	−4.8	33.4	190.0
6	18^h	187.4	22.7	188.8	26.8	−5.9	32.7	221.5

increase in velocity (ΔV) and the velocity (V) at the point where the ΔV is applied, according to the formula (Reference 7)

$$2V\,\Delta V = \frac{\mu}{a^2}\,\Delta a \qquad (2\text{-}1)$$

where μ is the product of the gravitational constant and the mass of the Earth. In order to make up a deficiency in the semimajor axis, or total energy, of the orbit, it is most efficient to apply ΔV at perigee. Since TTI is at perigee, the ΔV needed to correct a -3σ error would nearly equal 3σ or 17 meters/second, if it were applied at perigee. We must wait one orbit to do this, because the error is not known until the actual trajectory has been determined several hours after TTI. This is why the three-impulse strategy works. [The alternative would be to make a direct transfer by applying a midcourse correction as soon as possible after TTI.] Because the orbit must be determined, a new optimized transfer trajectory computed, and details of the midcourse maneuver calculated, the earliest that the maneuver can begin is about 18^h after TTI. By then, the spacecraft velocity is over five times smaller than at TTI, thus over five times the underburn error must be applied. The detailed calculations show that 93 meters/second are needed for the midcourse ΔV.

With a $V - 3\sigma$ error, the direct transfer ΔV costs are 128 meters/second, considerably lower than the 216 meters/second needed for the three-impulse strategy. But as the underburns at TTI increase, the direct transfer costs increases, while the three-impulse costs decrease. By $V - 6\sigma$, the direct costs are becoming prohibitive, being 222 meters/second, while the three-impulse costs are a more attractive 141 meters/second. For greater errors, the three-impulse strategy appears to be the most viable strategy.

Table 2-1 gives the direct transfer costs for an August 12, 1978, launch. Table 2-2 gives similar data for a July 23, 1978, launch, for which the costs are slightly greater, due to different lunar perturbations.

2.2 FIRST ORBIT MANEUVERS

When using the three-impulse contingency strategy, two additional maneuvers may be needed during the first orbit, to prevent Earth impact and/or ensure adequate ground tracking station coverage during the first perigee passage.

2.2.1 Period Change for Groundstation Coverage

Ground tracking stations for low-altitude parking orbits have been established near a great circle with inclination about 28 degrees and ascending node near the International Date Line, in order to optimize coverage for launches from Cape Canaveral. This network provides the data needed for relatively fast orbit determination and ensures a virtually continuous radio link with the spacecraft for the crucial transfer trajectory insertion maneuver. Because tracking station coverage is much poorer for other great circles, it is desirable to closely duplicate the parking orbit groundtrack during the low first orbit perigee passage, when the new TTI maneuver must be made. This will happen if the spacecraft orbital period is an integer number of sidereal days. Consequently, it will usually be necessary to change the period of the orbit by as much as 12 hours.

A change in period is equivalent to a change in semimajor axis according to Kepler's third law. The change can be made by changing the spacecraft velocity, where Equation (2-1), and the arguments for the direct transfer midcourse correction discussed in Section 2.1, apply. Using Kepler's third law, Equation (2-1) can be expressed in terms of the spacecraft orbital period, T:

$$\Delta V = \frac{\mu \Delta T}{3\, aVT} \qquad (2\text{-}2)$$

Table 2-2. Direct Transfer Costs for V-nσ for July 23, 1978 Launch

n	MC AT	MCC ΔV_{2XY} (m/sec)	MCC $\dot{Z}_2$ (m/sec)	MCC ΔV_2 (m/sec)	HOI ΔV_{IXY} (m/sec)	HOI $\dot{Z}_{IN}$ (m/sec)	HOI ΣV_{IN} (m/sec)	$T\Sigma V$ (m/sec)
3	18^h	96.8	10.0	97.3	40.9	−2.8	43.7	141.0
4	18^h	129.6	13.5	130.3	39.0	−3.8	42.8	173.1
5	18^h	162.8	17.2	163.7	35.4	−5.5	40.9	204.6
6	18^h	196.1	20.8	197.2	35.5	−6.0	41.4	238.6
3	24^h	108.7	11.2	109.3	39.7	−3.2	42.8	152.1
4	24^h	145.8	15.2	146.6	37.3	−4.3	41.7	188.2

Equation (2-2) can be used to compute the ΔV needed to cause a specified change in the period, ΔT . If this ΔV is applied, it will also change the semimajor axis by an amount Δa given by Equation (2-1) and the eccentricity by an amount Δe given by (Reference 8, p. 245):

$$\Delta e = 2\ (\cos f + e)\ \frac{\Delta V}{V} \qquad (2\text{-}3)$$

where f is the true anomaly at the point where the ΔV is applied.

A change in the radius of perigee, Δr_p , is given by:

$$\Delta r_p = (1 - e)\ \Delta a - a\Delta e \qquad 2\text{-}4$$

Equations (2-1), (2-3), and (2-4) can be combined to yield Δr_p as a function of ΔV:

$$\Delta r_p = a\left[\frac{2\ (1 - e)\ a\ V}{\mu} - \frac{2\ (\cos f + e)}{V}\right] \Delta V \qquad (2\text{-}5)$$

The ΔV costs computed with Equation (2-2) for period changes of $+1^h$ and $+12^h$, and the associated changes in a, e, and r_p , are listed in Table 2-3 for a maneuver performed 24^h after the attempted TTI, for five contigency cases. For errors algebraically less than -12σ , the costs for a 12^h period change become too large. Because the maneuver would probably be performed 18^h rather than 24^h after the attempted TTI, the velocity would ge greater, and thus the ΔV costs would be about 15-20 percent lower than given in the table. For the large errors (n greater than 12 for the $v - n\sigma$ cases), it becomes a matter of chance. If the orbit period is nearly an integer number of days, the maneuver can still be performed. Otherwise, it might be desirable to allow the spacecraft to complete two orbits before attempting the perigee maneuver. If one period was nearly an integer number of days plus 12^h, two periods would be nearly an odd integer number of days, and thus only a very small period change maneuver would be needed. In any case, the period change maneuver could be done near the first perigee, where it would be much more efficient, considerably reducing ΔV costs from those 24^h after perigee. For large n, the orbital period is small, because the line of apsides would not rotate too far in the RLP system during two orbits.

In order to test the validity of the two-body formulas, Equations (2-1) to (2-5), the ΔVs listed in Table 2-3 were applied 24^h after the launch and the spacecraft state propagated to first perigee using GMAS. The results are presented in Table 2-4. The actual change in the time of perigee ("actual ΔT" column) was 5 to 15 percent larger than expected. Since ΔV is proportional to ΔT, the actual ΔV costs to achieve period changes of 1^h and 12^h would run about 10 percent less than those given in Table 2-3, for the August 12, 1978 launch. Due to the nearly linear relation (as just noted, within 15 percent) between ΔV and ΔT, the actual ΔV needed to achieve a given ΔT can be found by rapidly convergent successive approximations. The small changes in the radius of perigee are within 100 km of the values computed with Equation (2-5), except for the $V - 5\sigma$ case with $\Delta T = 12^h$. Third-body perturbations are large enough to explain the differences. Values for the $V - 12\sigma$ and $V - 20\sigma$ cases are not listed in Table 2-3 since a 10 meter/second burn was needed at apogee in order to prevent atmospheric reentry at perigee, as described in Section 2.2.2. The apogee burn raises perigee by a greater amount than the period change maneuver 24^h after lanch.

A maneuver performed within one day of the attempted TTI (launch) will be designated ΔV_1 in the rest of this report. Certain cases require a ΔV_1 large enough to change the period by more than one day to avoid large lunar perturbations described in Section 2.4.2.

Table 2-3. ΔV Required at 24^h to Achieve Specified Change in Period, ΔT, and Associated Change in Orbital Elements, for Contingency Cases for August 12, 1978 Launch

n	a (km)	e	ΔT (hrs)	T (days)	V_{24^h} (m/sec)	ΔV (m/sec)	Δa (km)	Δe	$Δr_p$ (km)
5	318948	0.9775951	1	20.7501	1570.326	0.53	427.0	+0.0000154	+4.7
			12			6.39	5124.0	+0.0001856	+55.6
6	290165	0.9732738	1	18.0038	1542.856	0.69	447.7	+0.0000186	+6.6
			12			8.24	5370.7	+0.0002220	+79.1
9	228514	0.9709776	1	12.5824	1459.083	1.32	504.5	+0.0000122	+11.9
			12			15.84	6053.8	+0.0001462	+142.3
12	189549	0.9637666	1	9.5056	1373.226	2.24	553.9	–0.0000116	+22.3
			12			26.85	6646.9	–0.0001393	+267.2
20	130826	0.9486644	1	5.4505	1134.174	6.85	666.7	–0.0003667	+82.2
			12			82.14	8000.8	–0.0043972	+986.0

Table 2-4. GMAS Comparison With 2-Body Calculations

n	ATTEMPTED ΔT (hrs)	PERIGEE TIME	PERIGEE RADIUS (km)	GMAS ACTUAL ΔT	GMAS Δr_p (km)
5	0	SEP 02 $19^h 46^m$	7145.9		
	1	02 20 53	7114.5	$1^h 07^m$	−31.4
	12	03 9 29	6662.1	13 43	−483.0
6	0	AUG 30 22 45	7754.9		
	1	30 23 52	7763.4	$1^h 07^m$	+8.5
	12	31 12 20	7839.9	13 35	+85.0
9	0	AUG 25 9 14	6631.6		
	1	25 10 18	6650.3	$1^h 04^m$	+18.7
	12	25 22 37	6863.0	13 23	+231.4

2.2.2 Apogee Maneuver to Raise Perigee

The transfer trajectory insertion (TTI) was planned to be executed at a height of 184 km or a geocentric distance of 6560 km. The history of successive perigees of uncorrected orbits varies considerably with the size of the underburn at TTI, due to third-body perturbations. As an example, the perigee history for the $V - 6\sigma$ case for a July 23, 1978, launch is shown in Table 2-5. Fortunately, for most of the underburn cases, the third-body effects initially raise perigee. However, for some cases the perigee height decreases, leading to Earth impact at first perigee. For these cases, impact can be prevented by a small maneuver to increase the velocity at apogee, which raises perigee. Besides avoiding impact, it might also be useful to raise perigee to increase groundstation coverage during the perigee maneuver.

The change of perigee radius, Δr_p, is proportional to an impulse at apogee, ΔV_{ap}, according to the two body formula

$$\Delta r_p = C \, \Delta V_{ap} \tag{2-6}$$

where

$$C = 4 \sqrt{\frac{a^3}{\mu}} \sqrt{\frac{1-e}{1+e}} = \frac{4}{n} \frac{r_p}{r_{ap}} \tag{2-7}$$

with n being the spacecraft orbital mean motion, and r_p and r_{ap} being the radius at perigee and apogee, respectively, according to Reference 8. Values for a, e, and C for five contingency cases are given in Table 2-6. For the August 12, 1978, launch, values of C determined from GMAS calculations were 139.5 and 48.5 km/m/sec for the $V - 5\sigma$ and $V - 20\sigma$ cases, respectively. This

Table 2-5. Perigee History for V-6σ

PERIGEE HISTORY FOR V – 6σ

TIME		PERIGEE RADIUS (km)
DAY	HOUR, GMT	
78 AUG. 10	12	9,350
AUG. 30	13	14,078
SEP. 19	11	19.161
OCT. 7	3	24,082
OCT. 26	4	29,491
NOV. 13	21	25,883
NOV. 30	19	21,810
78 DEC. 19	6	17,326
79 JAN. 6	10	13,118
JAN 23	2	14,701
FEB. 10	9	17,418
FEB. 28	12	19,943
MAR. 16	23	28.514
APR. 4	13	36,184
APR. 23	0	36,158
79 MAY 10	2	42,196

Table 2-6. Change of Perigee Radius With Apogee Impulse ΔV_{ap}

1978 AUG. 12 LAUNCH FOR V–nσ CONTINGENCY CASES

n	a (km)	e	c
5	318,948	0.97759510	121.48
6	290,165	0.97327383	115.25
9	228,514	0.97097765	83.98
12	189,549	0.96376663	71.02
20	130,826	0.94866444	48.66

agrees to within 15 percent of the values in the table, 121.48 and 48.66 for these cases. Apogee impulses needed to avoid impact for the decreasing perigee cases were about 15 meter/second; specific values are given in Section 2.4.1.

2.3 PERIGEE MANEUVER

After completing one orbit, a maneuver must be performed near perigee to boost the spacecraft orbit's apogee to halo orbit distances. In this sense, it represents a new "launch" opportunity. Due to the large spacecraft velocity and the sensitivity of the transfer trajectory to errors, the maneuver must be carefully executed. The fact that perigee occurs in the Earth's shadow creates operational problems, and penalty factors caused by rapid spacecraft motion and finite burn duration must be computed.

2.3.1 Shadow Constraint

The spacecraft orbit lies close to the ecliptic and the spin axis is maintained perpendicular to the ecliptic plane. Consequently, changes in the spacecraft velocity are normally performed by using the radial jets in a pulsed mode. The jets are fired using signals generated by Sun sensors. While in the Earth's shadow, the pulses can be triggered instead with the help of the onboard clock.

A more serious problem is loss of power from the spacecraft's solar cells. Reliance on batteries for electric power while performing a maneuver would be risky. Consequently, it was decided to start the maneuver at exit from the Earth's penumbra rather than center the maneuver at perigee.

2.3.2 Penalty Factors for Finite Burns

The efficiency of a burn for increasing the energy of the orbit is proportional to the spacecraft velocity, according to Equation (2-1). The velocity is largest at perigee. As one moves away from perigee, the velocity decreases and the burn's efficiency decreases. For an impulsive ΔV, we can use two-body formulas to compute the efficiency, Eff_i , which is the ratio of the instantaneous velocity to the velocity at perigee:

$$Eff_i = \frac{(1 - e)(1 + e \cos E)}{(1 + e)(1 - e \cos E)} \qquad (2\text{-}8)$$

where E is the eccentric anomaly. $Eff_i = 1.0$ for $E = 0$ (perigee). Equation (2-8) must be integrated over the duration of the burn, and multiplied by the spacecraft acceleration imparted by the thrusters, to obtain the effective ΔV for boosting the energy of the spacecraft orbit. A 4-pound thruster imparts an acceleration of 0.03769 meter/second^2 and uses 0.008439 kg of hydrazine per second for a 471.74 kg spacecraft fully loaded with fuel (ISEE-3's planned mass). As fuel is used, the spacecraft weight decreases and the acceleration increases, according to the rocket equation. For the ΔVs in question, because a relatively small amount of fuel is used, the rocket equation was not required and the spacecraft mass was assumed constant. Thus, a 100 meters/second burn could actually be accomplished with a burn about 5 percent less, or with an efficiency 5 percent greater. Most contingency perigee burns are less than 100 meters/second and would have a proportionally smaller error due to neglect of the rocket equation. Fuel would also be used for earlier first-orbit maneuvers, as described in Section 2.2, decreasing the spacecraft weight and increasing the efficiency. For this reason, the efficiency varies with each case. In order to make a study of efficiencies which would be generally applicable, the rocket equation was not applied, so that the results are a few percent more pessimistic than most actual cases. The rocket equation is needed for accurate calculations for specific cases and is included in the engine model software used to compute details of ISEE-3 maneuvers.

Some operational advantage would be gained by using the radial jets. Since the spacecraft spin axis is maintained nearly perpendicular to the orbit plane, little or no reorientation would be needed for the maneuver. The jets would be fired in a pulse mode which could be timed to keep the vectored impulses in line with the velocity vector. In order to maintain the spacecraft attitude, the lower radial jets must thrust twice as often as the upper radials. Consequently, during two rotations of ISEE-3, the lower jet is fired twice while the upper jet is fired once during the maneuver. The jets are fired over an arc of 45 degrees, or one eighth of a rotation. Altogether, a radial jet is being fired 3/16th of the time. Since the jet fires over an arc of 45 degrees, there is an additional penalty factor since only the component parallel to the velocity vector contributes to the effective ΔV. The angle θ, measured from the jet to the velocity vector, ranges from $-\pi/8$ to $+\pi/8$ (± 22.5 degrees) while the jet is fired. The efficiency factor, Eff_θ, due to the size of the firing arc is given by the formula

$$Eff_\theta = \frac{8}{\pi} \int_{\frac{\pi}{8}}^{o} \cos \theta \, d\theta = \frac{\sin(\pi/8)}{\pi/8} = 0.9745 \qquad (2\text{-}9)$$

The acceleration imparted by the radial jets, a_r, is

$$a_r = \frac{3}{16} Eff_\theta \; (0.03769 \text{ meters/second}^2) = 0.006887 \text{ meters/second}^2 \qquad (2\text{-}10)$$

The effective ΔV using the radials, ΔV_r, as a function of time from perigee, t, can be calculated from Equations (2-8) and (2-10):

$$\Delta V_{r\,(t_2 - t_1)} = a_r \int_{t_1}^{t_2} Eff_i \,(t)\, dt \qquad (2\text{-}11)$$

Because Eff_i is defined in terms of E, the mean anomaly is computed from the orbital mean motion and the time from perigee, and E then computed by solving

Kepler's equation. Let ΔV_p be the impulsive ΔV which must be added to the spacecraft velocity at perigee. As long as ΔV_p is very small compared with the velocity at perigee, the burn can be considered to be symmetric about perigee. A 15-second step size was used in the summation to accomplish the integration. If the burn starts at penumbral exit, t_1 is set equal to the time of that event after perigee passage, and the summation is done until the entire ΔV_p is attained. The total amount of fuel for the radial maneuver, F_r, is then calculated:

$$F_r = \frac{3}{16}\,(0.008439 \text{ kg/sec})\,(t_2 - t_1) \tag{2-12}$$

The fuel efficiency, FE, is formed by dividing this by the desired ΔV_p for the effective perigee impulse:

$$FE_r = F_r/\Delta V_p \tag{2-13}$$

The fuel efficiency for a perfectly efficient burn would be:

$$FE_0 = \frac{\text{thruster fuel rate}}{\text{thruster acceleration}} = \frac{0.008439 \text{ kg/sec}}{0.03769 \text{ m/sec}^2} = 0.2239 \text{ kg/m/sec} \tag{2-14}$$

The penalty factor for the radial burn is:

$$PF_r = FE_r/FE_0 \tag{2-15}$$

and the ΔV penalty is

$$P_r = PF_r\,\Delta V_p \tag{2-16}$$

Fuel efficiencies and penalty factors were also computed for use of the axial jets. If the axials are used for the perigee maneuver, the spacecraft must be tilted about 90 degrees, so that the spacecraft axis is nearly aligned with the velocity vector. The burns are shorter than when using the radials since two axial thrusters can be fired continuously. The acceleration, a_a, is simply

$$a_a = 2\ (0.03769\ \mathrm{m/sec}^2) = 0.07538\ \mathrm{m/sec}^2 \qquad (2\text{-}17)$$

The lower axials would most likely be used. If for some reason it was necessary to use the upper axials, it would be necessary to multiply a_a by 0.6691, which is the cosine of 48 degrees, the angle by which the upper axials are canted to prevent plume impingement. Since the axials fire in a fixed direction, they can not follow the velocity vector like radials. If the spacecraft axis is aligned perpendicular to the orbital major axis, which would be optimum for a burn centered at perigee, the efficiency is degraded by the cosine of the instantaneous velocity vector to the spacecraft axis. This factor, $E_v(t)$, can be calculated from two-body formulas:

$$E_v(t) = \frac{\cos E - e}{1 - e \cos E} \qquad (2\text{-}18)$$

If the burn is started at penumbral exit, the axis can be aligned with the velocity vector at that point; then, E_v starts at 1.0 and slowly decreases as the direction of velocity changes. This is what has been done in the calculation of fuel efficiencies for the axials for burns started at penumbral exit. The burns are short enough that it would save very little to use an iterative optimization scheme to align the spacecraft axis with the mean direction of the velocity vector during the burn. An additional constraint is imposed by the heating characteristics of ISEE-3 which require the axis to be within 15 degrees of a plane perpendicular to the direction of the Sun. This constraint had little effect on the calculations.

The effective ΔV using axials, ΔV_a , is similar to Equation (2-11), but modified by the E_v factor:

$$\Delta V_{a\,(t_2 - t_1)} = a_a \int_{t_1}^{t_2} E_v\,(t)\,\mathrm{Eff}_i\,(t)\,dt \qquad (2\text{-}19)$$

ΔV_p is attained in the same way that it was for the radials described above. The total amount of fuel for the axial maneuver, F_a , is:

$$F_a = 2\ (0.008439\ \mathrm{kg/sec})\ (t_2 - t_1) \qquad (2\text{-}20)$$

The fuel efficiency, penalty factor, and ΔV penalties are computed using Equations (2-13), (2-15), and (2-16), where the subscript "r's" need to be changed to "a's."

Fuel efficiencies and penalties for three contingency cases (July 23, 1978, launch), illustrating values for different perigee radii are given in Table 2-7, with other pertinent quantities. The ratio of the penalty factors for penumbral exit versus perigee burns is given in the last column. It was decided that the axials would be used, since the long burn durations for the radials result in large penalty factors.

Penalties for the $V - 9\sigma$ case are listed in Table 2-8 for different tilts of the spacecraft axis from the perigee velocity vector direction and different burn start times. A 10-degree tilt was used to stay within the 15 degree limit imposed by the spacecraft heating constraint. A calculation was also done to see the effect of raising perigee by 6000 km for the $V - 9\sigma$ case. This resulted in a decreased perigee velocity, giving a penalty factor of 1.262 and a penalty of 15.4 meters/second over the 7000 km perigee radius case when the finite burns centered at perigee were computed.

Table 2-7. Fuel Rates for Finite Burns Near Perigee for V-nσ Contingency Cases

n	JETS	PERIGEE RADIUS (km)	IMPULSIVE ΔV_p (m/sec)	BURN CENTERED AT PERIGEE			PENUMBRAL EXIT		BURN STARTED AT PENUMBRAL EXIT			PENUMBRA / PERIGEE
				FUEL EFFICIENCY (kg/m/sec)	PENALTY FACTOR	DURATION, (min)	TIME FROM PERIGEE, (min)	TRUE ANOMALY	FUEL EFFICIENCY (kg/m/sec)	PENALTY FACTOR	DURATION, (min)	PENALTY FACTOR
3	AXIALS NOT CANTED	6579	20	0 225	1.005	4.5	13.9	63.5°	0.274	1.226	5.2	1.220
3	AXIALS CANTED 48°	6579	20	0.338	1 510	6.5	13.9	63.5°	0.421	1.880	8.2	1.246
3	RADIALS	6579	20	0 271	1 210	57.0	13.9	63.5°	0 405	1.809	85.0	1.494
6	AXIALS NOT CANTED	9345	35 68	0.225	1.005	8.0	15 2	45.9°	0.254	1.134	8.7	1.129
6	AXIALS CANTED 48°	9345	35 68	0 338	1 511	12 0	15 2	45.9°	0.390	1.744	13.5	1.154
6	RADIALS	9345	35.68	0.274	1 224	103.0	15.2	45.9°	0.390	1.740	146.0	1.422
9	AXIALS NOT CANTED	7186	53 4	0 229	1.024	12.0	11.2	49.3°	0.277	1.237	14.0	1.208
9	AXIALS CANTED 48°	7186	53 4	0.352	1.573	18.0	11.2	49.3°	0.442	1.974	23.0	1.255
9	RADIALS	7186	53.4	0.360	1.609	203 0	11.2	49.3°	0.61	2.72	260.0	1.69

Table 2-8. Finite Burn Penalty Factors for V-9σ Contingency Case

FINITE BURN*	BURN DURATION, (minutes)	PENALTY FACTOR	PENALTY (m/sec)
1. CENTERED AT PERIGEE†	12.2	1.026	1.4
2. START AT PERIGEE, IN LINE WITH PERIGEE VELOCITY	13.0	1.096	5.2
3. START AT END OF BURN #1, 6.^{m}1 FROM PERIGEE, IN LINE WITH PERIGEE VELOCITY	15.7	1.322	17.3
4. START AT PERIGEE, TILT 10° MORE FAVORABLE	12.5	1.058	3.1
5. START 6.^{m}1 FROM PERIGEE (=3), BUT TILT 10° MORE FAVORABLE	14.2	1.203	10.9

*ΔV PERIGEE = 53.7 m/sec (IMPULSIVE) USING AXIAL JETS, NOT CANTED; NOMINAL PERIGEE RADIUS = 7000 KM
†AT END, S/C VELOCITY IS 16° FROM PERIGEE VEL.

2.4 TRANSFER TRAJECTORY MIDCOURSE CORRECTIONS

The results of many calculations for different possible launch dates for the ISEE-3 spacecraft are presented in this subsection.

For most of the contingency case transfer trajectories, the total ΔV costs are only weakly dependent on the time of the second maneuver, which is performed after perigee. There is some advantage in doing the second maneuver early since the total ΔV costs are then less sensitive to the perigee maneuver execution errors. The Moon strongly perturbs some of the contingency transfer trajectories, either adding to (beneficial) or subtracting from (adverse) the energy of the spacecraft orbit.

2.4.1 Tables of Optimized Transfer Trajectories

The launch date for ISEE-3 was originally set at July 23, 1978. The date was selected so that the Moon, the Sun, and the Earth would be well-separated as seen from the spacecraft during the early part of the transfer flight. Consequently, geometry would be favorable for attitude determination. The Sun-Earth-Moon angle was 137 degrees and decreasing (Moon waning). Our contingency study effort concentrated on the July 23rd date until the launch was postponed.

The results for July 23rd are given in Table 2-9. Some of the July 23rd cases were the most thoroughly examined and optimized; they provided the experience needed to more quickly optimize the cases for other launch dates.

On the newly-scheduled August 12, 1978, launch date, the Sun-Earth-Moon angle was 108 degrees and increasing. With the Moon waxing at launch rather than waning, the lunar perturbations of the various contingency case trajectories would be very different for the two launch dates, and thus a new study was begun for the August 12th launch. Because this was the prime launch date, more contingency cases were studied for it than for any other date. The results are presented in Table 2-10. A study was also performed for an August 13th launch,

Table 2-9. Fuel-Optimum Contingency Strategies for V -nσ July 23, 1978 Launch

MIDCOURSE															
n	$\Delta V_{20}{}^{h}$ (m/sec)	ΔV_{ap} (m/sec)	PERIOD (days)	APOGEE (km)	PERIGEE (km)	ΔV_p (m/sec)	TIME (days)	$\dot{z}$ (m/sec)	ΣV_{MC} (m/sec)	$\dot{z}_{IN}$ (m/sec)	ΣV_{HOI} (m/sec)	TOTAL ΣV (m/sec)	TOTAL TIME (days)	1978 PERIGEE DATE, TIME	R_e (km)
														m d h	
3	0	0	29.6	791,612	6,579	20.0	10	−14.9	120.0	0.2	72.8	212.8	117.8	8 22 6	361,180
4	0	8	24.3	701,663	7,332	21.0	5	12.2	74.3	20.4	92.8	196.1	117.6	8 17 1	306,057
5	0	0	20.6	630,953	7,514	27.5	44.1*	−2.5	21.3	13.3	108.9	157.8	114.6	8 13 7	356,933
6	0	0	17.8	573,138	9,345	35.68	85	9.6	13.3	1.1	92.7	141.8	112.7	8 10 12	361,198
7	35	0	17.7	570,463	9,805	36.6	46.2*	5.1	6.5	11.4	101.5	179.6	112.6	8 10 9	361,204
8	0	30	14.1	484,088	6,504	47.8	48.3*	20.2	28.7	5.1	77.6	184.1	113.6	8 6 19	88,920
9	0	24	12.6	449,144	7,186	53.4	5	−4.2	4.8	3.6	71.1	153.3	111.5	8 5 7	163,185
10	0	15	11.3	418,887	7,290	58.4	49.6*	−2.6	4.3	6.8	70.1	147.8	110.4	8 4 1	225,226
11	0	0	10.3	392,434	6,509	60.0	5	14.6	19.3	15.5	72.4	151.7	110.5	8 3 0	179,894
12	0	0	9.4	369,107	6,701	65.1	5	26.4	31.8	17.5	63.6	160.3	110.1	8 2 3	133,441

*MIDCOURSE PERFORMED AT APOGEE.

Table 2-10. Fuel-Optimum Contingency Strategies for V -nσ August 12, 1978 Launch

MIDCOURSE															
n	$\Delta V_{20}{}^{h}$ (m/sec)	ΔV_{ap} (m/sec)	PERIOD (days)	APOGEE (km)	PERIGEE (km)	ΔV_p (m/sec)	TIME (days)	$\dot{z}$ (m/sec)	ΣV_{MC} (m/sec)	$\dot{z}_{IN}$ (m/sec)	ΣV_{HOI} (m/sec)	TOTAL ΣV (m/sec)	TOTAL TIME (days)	1978 PERIGEE DATE,TIME	R_e (km)
														m d h	
3	0	0	30.1	797,109	11,800	21.1	41.8*	−3.6	21.4	16.7	173.6	216.1	121.0	9 11 19	351,420
4	0	0	24.9	703,440	17,725	24.0	42.9*	23.9	25.3	35.4	205.9	255.2	116.2	9 6 12	77,358
5	0	0	21.1	630,749	7,146	30.6	45.8*	−1.0	1.3	11.4	131.4	163.3	115.8	9 2 20	172,213
6	0	0	18.3	572,575	7,755	34.6	43	−8.9	13.8	0.0	92.8	141.2	112.5	8 30 23	137,953
7	30	0	17.9	564,753	7,890	35.0	45.2*	−11.4	17.5	−6.0	88.1	170.7	112.1	8 30 14	116,667
8	−15	6	12.9	454,680	7,140	55.2	43.9*	−14.6	20.1	−3.5	112.9	209.2	109.1	8 25 14	253,458
9	0	0	12.7	450,395	6,632	53.7	44.6*	−4.2	7.2	6.0	110.1	171.0	110.0	8 25 9	276,846
10	0	5	11.5	420,840	6,799	59.3	44.6*	−2.4	7.4	8.7	99.8	171.5	109.0	8 24 4	367,190
12	10	10	9.6	372,230	6,868	66.3	1	3.7	36.9	8.1	72.1	185.3	109.0	8 22 6	367,325
20	0	10	5.5	254,935	6,716	109.7	7	−0.4	1.2	6.0	57.4	178.3	106.9	8 18 3	353,204

*MIDCOURSE PERFORMED AT APOGEE.

in case of a 1-day postponement; the results are given in Table 2-11. No study was performed for the alternate August 21st launch date, since the Sun-Earth-Moon angle was then 142 degrees with the Moon waning. The geometry and lunar perturbations would be similar to those for the July 23rd launch.

The total ΔV budget of the ISEE-3 spacecraft was about 450 meters/second. Because at least 100 meters/second was planned for halo orbit stationkeeping and attitude maneuvers, it was hoped that transfer trajectory and halo orbit insertion costs could be held under 300 meters/second (and prefereably much less, to conserve fuel for a possible extended mission). The results tabulated in this section do not include penalty factors due to finite burns and performance of the TTI at penumbral exit, rather than at perigee (see Section 2.3). Correction of execution errors, especially for the perigee maneuver, could be significant and are not estimated in the table. Consequently, 200 meters/second was selected as an upper limit for an acceptable contingency case budget (TΣV in the tables).

The expected mean error, σ , of the velocity, V , at transfer trajectory insertion was 5.6 meters/second. The velocity, v , at TTI for a given contingency case was defined by the equation

$$v = V - n\sigma \quad (2\text{-}21)$$

where n is a number greater than 3 for a contingency situation. The different contingency cases can conveniently be specified by the value for n , given in the first column of the table. All velocities are given in meters per second in the tables. All distances are in kilometers, and times are in mean solar days. For some cases, the period of the spacecraft orbit needs to be changed by an impulse in the direction of the velocity vector 20 hours after launch, as described in Section 2.2.1 and 2.4.3. The impulse is given under the $\Delta V_{20}h$ column. In order to facilitate the study, phasing maneuvers to obtain optimum

Table 2-11. Fuel-Optimum Contingency Strategies for V -nσ August 13, 1978 Launch

MIDCOURSE															
n	$\Delta V_{20}{}^h$ (m/sec)	ΔV_{ap} (m/sec)	PERIOD (days)	APOGEE (km)	PERIGEE (km)	ΔV_p (m/sec)	TIME (days)	$\dot{z}$ (m/sec)	ΣV_{MC} (m/sec)	$\dot{z}_{IN}$ (m/sec)	ΣV_{HOI} (m/sec)	TOTAL ΣV (m/sec)	TOTAL TIME (days)	1978 PERIGEE DATE, TIME	R_e (km)
														m d h	
5	0	5	21.2	632,052	6,668	30.9	46.1*	4.6	5.9	10.8	125.8	167.6	116.3	9 3 21	315,967
6	0	0	18.3	573,713	7,814	35.7	45.5*	−6.2	10.4	7.0	106.3	152.4	112.9	8 31 24	188,942
7	15	0	17.0	544,920	7,871	37.2	45.7*	−12.5	17.5	−7.1	85.4	155.1	111.5	8 30 15	114,480
8	0	0	14.2	485,418	7,288	51.6	45.0*	−17.3	18.1	−21.3	145.6	235.3	110.7	8 27 22	116,618
9	15	0	12.2	438,559	6,756	56.4	44.7*	−6.1	9.2	3.5	108.4	189.1	109.5	8 25 21	246,923
10	0	0	11.5	421,389	6,723	59.1	44.4*	−4.7	12.8	5.6	99.8	171.7	108.9	8 25 4	294,254
12	0	10	9.6	372,592	7,129	71.2	45.1*	−2.4	7.4	7.8	85.7	174.4	107.9	8 23 6	369,278
20	0	10	5.4	255,005	6,776	110.6	7.0	2.4	5.3	8.0	59.8	185.7	106.7	8 19 3	355,540

*MIDCOURSE PERFORMED AT APOGEE.

station coverage at perigee have not been calculated. Only period changes needed to avoid unfavorable lunar perturbations discussed in Section 2.4.3 have been investigated. The velocity at apogee occasionally needs to be increased by an amount ΔV_{ap} given in the third column. The purpose is to avoid Earth impact (or atmospheric reentry) as discussed in Section 2.2.2. The period, the apogee distance, and the perigee distance of the first orbit are specified in the next three columns. ΔV_p is the impulse applied at perigee to increase the energy of the spacecraft orbit to reach L_1 distances, as discussed in Section 2.3. The second maneuver (midcourse correction) is specified in the next three columns. The first of these is the time from perigee; selection of this time is discussed in the next section. The spacecraft is maintained with its axis perpendicular to the ecliptic plane, with the midcourse correction (MCC) resolved into in-plane (ΔV_{XY}) and out-of-plane ($\dot{Z}$) components to be executed by the radial and axial jets, respectively. The sum of ΔV_{XY} and $\dot{Z}$ is ΣV_{MCC}. The quantity ΔV_{XY} is not tabulated, but can be calculated from ΣV_{MCC} and $\dot{Z}$. The halo orbit insertion maneuver, performed when the spacecraft crosses the $Y = 0$ plane in the RLP reference frame, is specified in the next two columns. The out-of-plane ($\dot{Z}_{IN}$) component and sum (ΣV_{HOI}) are given, similar to the midcourse correction.

The total ΔV used for the contingency case is given under the total ΣV column. It is the variable which is optimized (minimized) and is the sum of $\Delta V_{20}h$, ΔV_{ap}, ΔV_p, ΣV_{MCC}, and ΣV_{HOI}. The total flight time required, from launch to halo orbit insertion, is given in the next column. The 1978 month, day and hour (Greenwich Mean Time) of perigee are listed next. The distance of closest approach to the Moon is given in the last column.

2.4.2 Optimization Strategy

Software was designed for use with the GMAS to perform the calculations needed to construct Tables 2-9, 2-10, and 2-11. A subroutine was written to apply an impulsive ΔV in the direction of the velocity vector for modeling $\Delta V_{20}h$,

ΔV_{ap} , and ΔV_p . TRCOWL, the time-regularized Cowel integration subroutine, was modified to define a stopping condition when the spacecraft crossed the $Y = 0$ plane in RLP coordinates. Subroutines were written to define three independent variables to be the components of an impulsive ΔV vector (the midcourse correction) and to define two dependent variables and one optimization variable. The two dependent variables were the values of X and Z (RLP coordinates), which must be 239267.95 km and -100326.75 km, respectively, for the desired halo orbit (these are the RLP coordinates of the HOI point for $Y_{RLP} = 0$). The optimization variable was defined to be the sum of the absolute values of the in-plane (ecliptic) and out-of-plane components of the impulsive ΔV for the midcourse correction and for the halo orbit insertion. A further optimization of the total ΔV was accomplished by manually varying ΔV_p and the time of the midcourse correction. Automation of this last step was undesirable due to the excessive computer time that would be needed and consequent slow turnaround. In some cases, multiple solutions existed, so that unfavorable cases might be selected by an automatic procedure. Care was taken to avoid hyperbolic orbits which did not pass close to the halo orbit.

The most important aspect of optimization is selection of ΔV_p . With the proper ΔV_p , the spacecraft trajectory will reach the HOI point in the X-Y plane, as shown by the solid curve in Figure 2-4 (RLP coordinates are used). Such a trajectory requires no ecliptic-plane midcourse correction, but an out-of-plane maneuver is needed to achieve $Z = 100326.75$ km at HOI. If ΔV_p is too small, the transfer orbit will not have enough energy and will intersect the $Y = 0$ plane short of HOI, at point A in Figure 2-4. With ΔV_p too large, the trajectory reaches the $Y = 0$ plane at some point B beyond the HOI point, and the spacecraft will escape into a heliocentric orbit.

The optimization of ΔV_p is illustrated in Table 2-12, where the midcourse correction is performed at a fixed time, 5 days after perigee. The MCC data are given in columns 2-4 (ΔV_{2XY} = ecliptic plane component of the ΔV , $\dot{Z}_2$ is the ecliptic normal component, and $\Sigma V_2 = \Delta V_{2XY} + \dot{Z}_2$), similar data are given

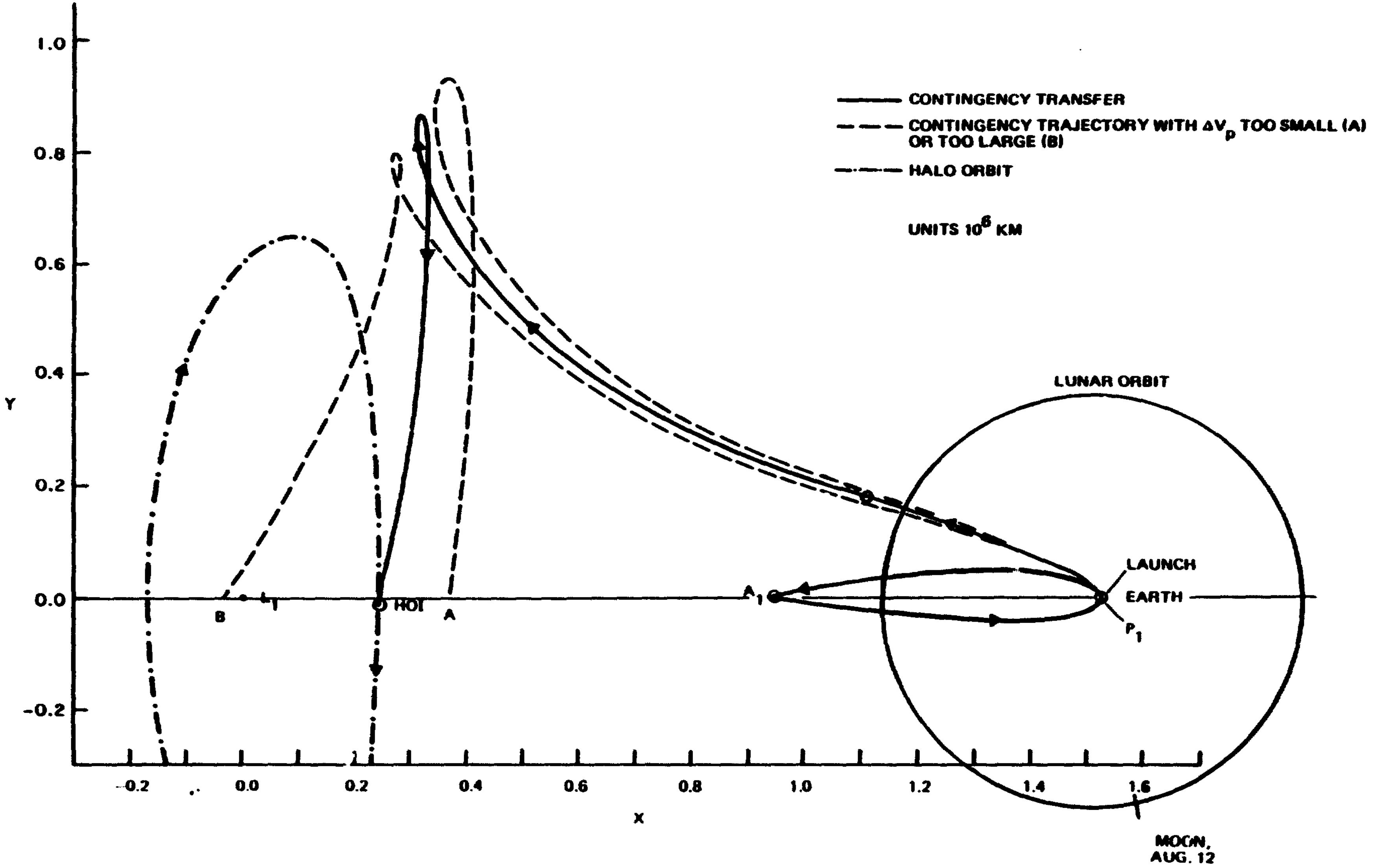

Figure 2-4. V-6σ Contingency Case, Launch August 12, 1978, with ΔV_p Errors

Table 2-12. Optimization of V-9σ Case for July 23, 1978, Launch

ΔV_p (m/sec)	ΔV_{2xy} (m/sec)	$\dot{z}_2$ (m/sec)	ΣV_2 (m/sec)	ΔV_{IXY} (m/sec)	$\dot{z}_{IN}$ (m/sec)	ΣV_{IN} (m/sec)	$T\Sigma V$ (m/sec)
53.2	2.9	−3.7	6.6	68.7	3.8	72.5	156.3
53.3	1.1	−3.9	5.1	67.9	3.7	71.6	154.0
53.4	0.6	−4.2	4.8	67.5	3.6	71.1	153.3←
53.5	2.3	−4.4	6.7	66.8	3.5	70.3	154.5
53.6	4.0	−4.7	8.7	66.1	3.5	69.6	156.9
58.63*	1.3	−3.0	4.3	66.4	3.6	70.0	156.9

*PERIGEE BURN AT PENUMBRAL EXIT, NOT PERIGEE for HOI in the next three columns (ΔV_{IXY}, Z_{IN}, ΣV_{IN}), and the total ΔV required is given under $T\Sigma V = \Delta V_p + \Sigma V_2 + \Sigma V_{IN}$. As expected from the above discussion, the minimum $T\Sigma V$, marked by an arrow, occurs when ΔV_{2XY} is near zero. Since $\dot{Z}_2$ changes slowly with ΔV_p, ΣV_2 is also optimized. The HOI costs continue to decrease as ΔV_p increases.

Table 2-13 shows the minimum $T\Sigma V$ found when the MCC was performed at several different times from perigee, for the $V - 6\sigma$ case with a July 23, 1978 launch. The time from perigee is given in the first column. As noted above, the quality of the optimization of $T\Sigma V$ can be approximately gaged by the smallness of ΔV_{2XY}. It is clear from the table that the value of the minimized $T\Sigma V$ varies little with time of MCC from perigee. The values vary by only a few percent, except near 20 days, when about 20 meters/second additional ΔV is required. The reason is that an approximately 20 meters/second out-of-plane ΔV is needed at HOI for these cases, while usually this $\dot{Z}_{IN}$ is only a few meters/second. The MCC $\dot{Z}$ is also somewhat larger. At 20 days, the spacecraft reaches its maximum height above the ecliptic plane, while near HOI, it is far below the plane, near its minimum value. The small plane change needed to reach the HOI point is more economically accomplished while the spacecraft is relatively close to the ecliptic.

The last two columns of Tables 2-3 give values for ΔV_{2XY} and the difference in $T\Sigma V$ from the minimum value give in the 4th column, when ΔV_p is 0.01 meter/second larger or smaller than the ΔV_{per} value for the minimum $T\Sigma V$. There is virtually no differences when MCC is performed soon after perigee, but the values increase considerably as the time of MCC increase. This stronger dependence on ΔV_p makes optimization of the cases at later times from perigee easier than the early cases. Therefore, it was desirable to do the MCC at apogee, which occurred at 46.0 days for the $V - 6\sigma$ case. For many of the cases listed in Tables 2-9, 2-10, and 2-11, the MCC was performed at apogee only, to facilitate the study. In practice, it would be best

Table 2-13. Minimum at Given Times, Varying ΔV_p

TIME (DAY)	ΔV_p (m/sec)	MINIMUM (m/sec)		VALUES FOR $\Delta V_p \pm .01$ (m/sec)	
		ΔV_{2XY}	$T\Sigma V$	ΔV_{2XY}	$T\Sigma V - MIN$
1	35.68	1.8	150.5		
2	35.76	1.0	147.0	1.0	0.0
3	35.74	1.0	145.7	1.1	0.0
4	35.68	1.5	146.2		
5	35.68	1.3	151.3		
20	35.63	1.6	164.6	1.1	0.1
45	35.68	0.1	147.0		
46	35.68	0.2	146.8	0.7	0.3
47	35.68	0.2	146.6		
48	35.68	0.2	146.3		
50	35.68	0.2	146.0	0.9	0.5
55	35.68	0.2	145.2		
60	35.68	0.3	144.3		
70	35.68	0.8	143.3	7.1	2.2
80	35.68	2.2	142.2	10.2	0.7
85	35.68	3.7	141.8	16.2	0.6
90	35.6785	1.8	141.7	1.6	24.2

to perform the MCC soon after perigee, since $T\Sigma V$ is then less sensitive to errors in ΔV_p . Energy considerations (Equation 2-1) and Figure 2-4 show that it is desirable to correct execution errors in ΔV_p as soon as possible.

The dependence of $T\Sigma V$ on ΔV_p is shown in more detail for four MCC times in Table 2-14. If the MCC is performed during the early or late stages of the transfer trajectory, optimization does not occur exactly when ΔV_{2XY} vanishes due to rapidly-changing ecliptic plane ΔV costs at HOI. The HOI ΔV costs can be significantly decreased by increasing the MCC, while $T\Sigma V$ changes little. At 90 days from perigee (only 5 days before HOI), there are multiple solutions, as listed in Table 2-15.

2.4.3 Lunar Perturbations

The major effect of the Moon is a change in the energy of the spacecraft orbit, described in Section 2.4.3.1. Large ΔVs normal to the ecliptic, needed to correct a change in the inclination of the transfer trajectory plane caused by relatively close approaches to the Moon, are discussed in Section 2.4.3.2.

2.4.3.1 Change in Spacecraft Orbital Energy

If the spacecraft-Earth-Moon angle is less than 60 degrees when the spacecraft crosses the Moon's orbit on its transfer trajectory to the halo orbit, the spacecraft orbital energy will be substantially modified by lunar gravity. If the spacecraft passes in front of the Moon (i.e., it arrives at the crossing point before the Moon), its orbital energy will be increased, and thus a smaller ΔV_p , and smaller $T\Sigma V$, is needed to reach the halo orbit. But if the spacecraft arrives later, passing behind the Moon, the orbital energy is decreased, and a much larger ΔV is needed at perigee to compensate for the loss.

The Moon crossed a halo-orbit-bound contingency transfer trajectory on August 2 and September 1, 1978. These were 9-1/2 and 28-1/2 days after the July 23rd launch, respectively. Since it takes 2-1/2 days for the spacecraft to reach the lunar orbit from perigee, the critical contingency orbit periods were

Table 2-14. Some Details of Optimization of the V -6σ Case

ΔV_p (m/sec)	AT 2^d (m/sec)		AT 20^d (m/sec)		APOGEE ($46.^d0$) (m/sec)		AT 70^d (m/sec)	
	ΔV_{2XY}	TΣV	ΔV_{2XY}	TΣV	ΔV_{2XY}	TΣV	ΔV_{2XY}	TΣV
35.63			1.6	164.6←				
35.64			1.1	164.7				
35.65			0.6	164.7				
35.66	1.7	148.0	0.1	164.9	1.3	147.4	5.8	146.8
35.67			0.4	165.1				
35.68			1.0	165.4	0.2	146.8←	0.8	143.3←
35.70	1.2	147.2			1.6	148.8	7.1	145.5
35.72					3.1	150.7		
35.74	1.1	147.0			4.6	152.8	19.1	150.0
35.76	1.0	147.0←						
35.78	0.9	147.0						

Table 2-15. Optimization for MCC 90^d After Perigee, V -6σ Case

EQUATORIAL COMPONENT STARTING VALUES										
ΔV_{2X} (m/sec)	ΔV_{2Y} (m/sec)	ΔV_{2Z} (m/sec)	ΔV_p (m/sec)	ΔV_{2XY} (m/sec)	$\dot{Z}_2$ (m/sec)	ΣV_2 (m/sec)	ΔV_{IXY} (m/sec)	$\dot{Z}_{IN}$ (m/sec)	ΣV_{IN} (m/sec)	$T\Sigma V$ (m/sec)
5.0 or	5.0	5.0	35.677	0.6	19.1	19.7	94.8	−8.6	103.4	158.8
−0.042	−1.131	20.236	35.6785	3.4	19.1	22.5	91.8	−8.6	100.4	158.6 ←
5.0	5.0	5.0	35.68	7.0	19.0	26.0	88.4	−8.6	97.0	158.8
−0.042	−1.131	20.236	35.68	7.2	19.0	26.2	88.2	−8.6	96.9	158.8
1.552	9.464	−21.358	35.677	0.4	9.6	9.9	94.8	1.4	92.6	141.8
−1.552	9.464	−21.358	35.6785	1.8	9.6	11.4	93.4	1.3	94.7	141.7 ←
−1.552	9.464	−21.358	35.68	3.8	9.6	13.4	91.5	1.1	96.2	141.8

7 days and 36 days. None of the orbits listed in Table 2-9 have periods very close to these values, although the V - 12σ case, with a 9.4 day period, is affected. The increasing $\dot{Z}$ (out-of-plane) costs, discussed in the next subsection, provide a clue to the strength of the lunar perturbations. The critical contingency orbit must be near V - 15σ. Between V - 11σ and V - 15σ, the Moon is adding energy to the orbit. Contingency cases worse than V - 15σ would have been exacerbated by unfavorable lunar perturbations and would have been very difficult to salvage. A close approach to the Moon would have occurred on the inward bound leg of the first orbit for the V - 7σ and V - 8σ cases. For the V - 8σ case, a large apogee burn was needed to prevent Earth impact, since the lunar perturbations decreased the perigee distance (smaller apogee burns would have been needed for the V - 9σ and V - 10σ cases, for the same reason). For the V - 7σ case, a large burn 20 hours after launch would have been needed to avoid severe lunar perturbations. The ΔV at 20^h effectively changed the case to a V - 6σ case. This strategy was examined in more detail for the August 12th launch cases.

For the August 12th waxing-Moon launch, a close approach to the Moon occurred on the inbound leg of the first orbit for the V - 4σ case. From Table 2-10, we see that the Moon would have nearly tripled the perigee height, and necessitated large out-of-plane maneuvers. Comparison with Table 2-1 shows that direct transfers are more economical in this case, and are preferred. At V - 5σ, the lunar perturbations are much smaller and the contingency strategy becomes more efficient than a direct transfer.

A calculation like the one described above for the July 23rd launch shows that, for the August 12th launch, the critical orbital period is about 16 days for lunar perturbations of the transfer trajectory. This is the period of the first orbit for the V - 7σ case. The V - 6σ case, where the spacecraft would have passed in front of the Moon, was greatly aided by the lunar attraction, and thus the total maneuver ΔV costs are lower than for any of the other contingency cases.

The rapid increase in ΔV costs are shown in Table 2-16, where contingency cases at $1/4\sigma$-intervals are tabulated from $V - 6\sigma$ to $V - 7\text{-}3/4\sigma$, with no first-orbit maneuvers. Closest approach to the Moon occurred for $V - 6\text{-}3/4\sigma$ (70,200 km distant). Most of the increased ΔV costs for the surrounding cases are out-of-plane corrections, but the largest total ΣV, an unacceptable 254.6 meters/second, occurred for the $V - 7\sigma$ case, with the spacecraft passing close to the Moon, and behind it.

The problems illustrated in Table 2-16 can be ameliorated by changing the period of the first orbit with a maneuver performed 20^h after launch, similar to the maneuver described in Section 2.2.1, but larger. For most of the cases listed in Table 2-16, it was found that increasing the first-orbit period to the value for the $V - 6\sigma$ case gave optimum results. These as shown in Table 2-17. For larger n, ΔV_{20^h} had to be increased to achieve the $V - 6\sigma$ orbit. The total ΣV costs, which include ΔV_{20^h}, were reduced even for the $V - 7\text{-}3/4\sigma$ case, one full σ beyond the uncorrected closest lunar approach case, where 55 meters/second were used to reach the $V - 6\sigma$ orbit. Only for the $V - 8\sigma$ case was it more economical to decrease the spacecraft orbit period, so that it become similar to the $V - 9\sigma$ case. The costs to correct the $V - 8\sigma$ orbit are still unacceptably high. A delayed transfer, described in Section 2.5, might be preferred for such a case. The actual ΔV applied at 20^h would usually have to be changed slightly from the value given in Table 2-17 to achieve good station coverage at perigee, which must occur at about the same time of day as launch, as described in Section 2.2.1. Fortunately, the total cost is not greatly affected by this. For example, several ΔV_{20^h} values were used for the $V - 7\sigma$ case, producing perigee times at different hours of August 30, 1978. For a perigee time range of 15 hours centered on the time producing the minimum $T\Sigma V$, the $T\Sigma V$ was less than 2 meters/second more than the minimum. For a range of 28 hours, the $T\Sigma V$ difference was under 8 meters/second.

Table 2-16. Fuel-Optimum Contingency Strategies for V-6σ to V-8σ, August 12, 1978 Launch. No First-Orbit Maneuvers

n	ΔV_{20^h} (m/sec)	ΔV_{ap} (m/sec)	PERIOD (days)	ΔV_p (m/sec)	$\dot{Z}_{mc}$ (m/sec)	ΣV_{mc} (m/sec)	$\dot{Z}_{in}$ (m/sec)	ΣV_{in} (m/sec)	TOTAL ΣV (m/sec)	PERIGEE DATE, TIME			R_m (km)
										m	d	h	
6	0	0	$18^{d}.3$	34.6	− 8.9	13.8	0.0	92.8	141.2	8	30	23	137,953
6¼	0	0	18.0	34.9	−13.6	17.9	−11.0	94.7	147.5	8	30	16	105,900
6½	0	0	17.1	35.8	−23.6	27.8	−34.6	114.4	178.0	8	29	18	87,000
6¾	0	0	16.5	39.4	−33.3	34.4	−58.6	152.3	226.1	8	29	5	70,200
7	0	0	16.0	44.5	−29.9	35.4	−53.1	174.6	254.6	8	28	17	82,850
7¼	0	0	15.5	47.0	−21.1	36.4	−31.5	153.4	236.7	8	28	5	95,500
7½	0	0	15.1	48.4	−15.0	20.9	−17.0	142.6	211.9	8	27	18	123,500
7¾	0	0	14.7	49.3	−11.1	16.0	− 7.8	129.9	195.2	8	27	7	149,400

Table 2-17. Fuel-Optimum Contingency Strategies for V-6σ to V-8σ, August 12, 1978 Launch

MIDCOURSE															
n	$\Delta V_{20}{}^{h}$ (m/sec)	ΔV_{ap} (m/sec)	PERIOD (days)	APOGEE (km)	PERIGEE (km)	ΔV_p (m/sec)	TIME (days)	$\dot{z}$ (m/sec)	ΣV_{MC} (m/sec)	$\dot{z}_{IN}$ (m/sec)	ΣV_{HOI} (m/sec)	TOTAL ΣV (m/sec)	TOTAL TIME (days)	1978 PERIGEE DATE, TIME	R_e (km)
														m d h	
6	0	0	18.3	572,575	7,755	34.6	43	−8.9	13.8	0.0	92.8	141.2	112.5	8 30 23	137,953
6¼	5	0	18.0	566,966	7,755	34.8	45.4*	−10.4	13.2	−3.5	92.6	145.6	112.4	8 30 11	123,297
6½	10	0	17.7	561,439	7,750	35.0	46.0*	−12.8	13.9	−9.5	95.3	154.2	112.4	8 30 10	109,444
6¾	20	0	17.8	563,103	7,820	35.0	45.6*	−12.1	14.6	−7.8	91.8	161.4	112.3	8 30 12	113,040
7	20	0	17.9	564,753	7,890	35.0	45.2*	−11.4	17.5	−6.0	88.1	170.7	112.1	8 30 14	116,667
7¼	40	0	18.0	566,389	7,959	35.0	44.8*	−10.9	20.4	−4.4	84.6	180.0	111.9	8 30 16	120,317
7½	45	0	17.7	560,717	7,955	35.3	46.4*	−13.6	15.2	−11.6	91.8	187.2	112.7	8 30 10	106,235
7¾	55	0	17.8	562,271	8,027	35.3	46.0*	−12.8	14.9	−10.0	88.1	193.4	112.5	8 30 12	109,507
8	−15	6	12.9	454,680	7,140	55.2	43.9*	−14.6	20.1	−3.5	112.9	209.2	109.1	8 25 14	253,458

*MIDCOURSE PERFORMED AT APOGEE.

2.4.3.2 Inclination Changes

Part of the Moon's orbit, and the first orbit for the $V - 6\sigma$ contingency case. (August 12, 1978 launch), are shown in Figure 2-5. Both orbits are inclined about 6 degrees to the ecliptic, but the ascending nodes are separated by about 172 degrees. As a result, the mutual inclination of the planes exceeds 10 degrees. Since the spacecraft orbit is highly elliptical, the points where the spacecraft and lunar distances are equal are located near apogee in the geocentric view of Figure 2-5. Because this is about 40 degrees from the common node, the minimum separation can be no less than about 50,000 km. A ΔV of about 100 meters/second normal to the ecliptic is needed to change the spacecraft plane enough to encounter the Moon, or to make a significant lunar swingby maneuver. Such a ΔV would have been too large; avoidance of the Moon, as described in the previous subsection, was found to be the best contingency strategy. Segments of the transfer trajectories for the $V - 6\sigma$ and $V - 6.75\sigma$ (with no $\Delta V_{20}h$) cases are shown in Figure 2-4. The difference between the two segments illustrates the strong perturbation for the $V - 6.75\sigma$ case, where the spacecraft passes about 70,000 km above the Moon. This difference must be corrected by large out-of-plane ΔVs which make the total costs too large. For other launch dates, the first spacecraft orbit maintains about the same orientation with respect to the Sun, so that the inclination to the ecliptic remains about 6 degrees, and the longitude of the ascending node is about 120 degrees less than that of the Sun at launch. Since the motion of the node of the lunar orbit is much slower than the motion of the Sun, different geometries occur for launches at different times of the year. The July 23rd launch geometry was similar to that for the August 12th launch; the closest possible approaches to the Moon were slightly greater for the July launch than for an August launch. Therefore, the remarks in the previous paragraph are applicable to the July 23rd contingency cases. For launch dates a few months earlier or later than those considered here, the lunar perturbations would be very different, and

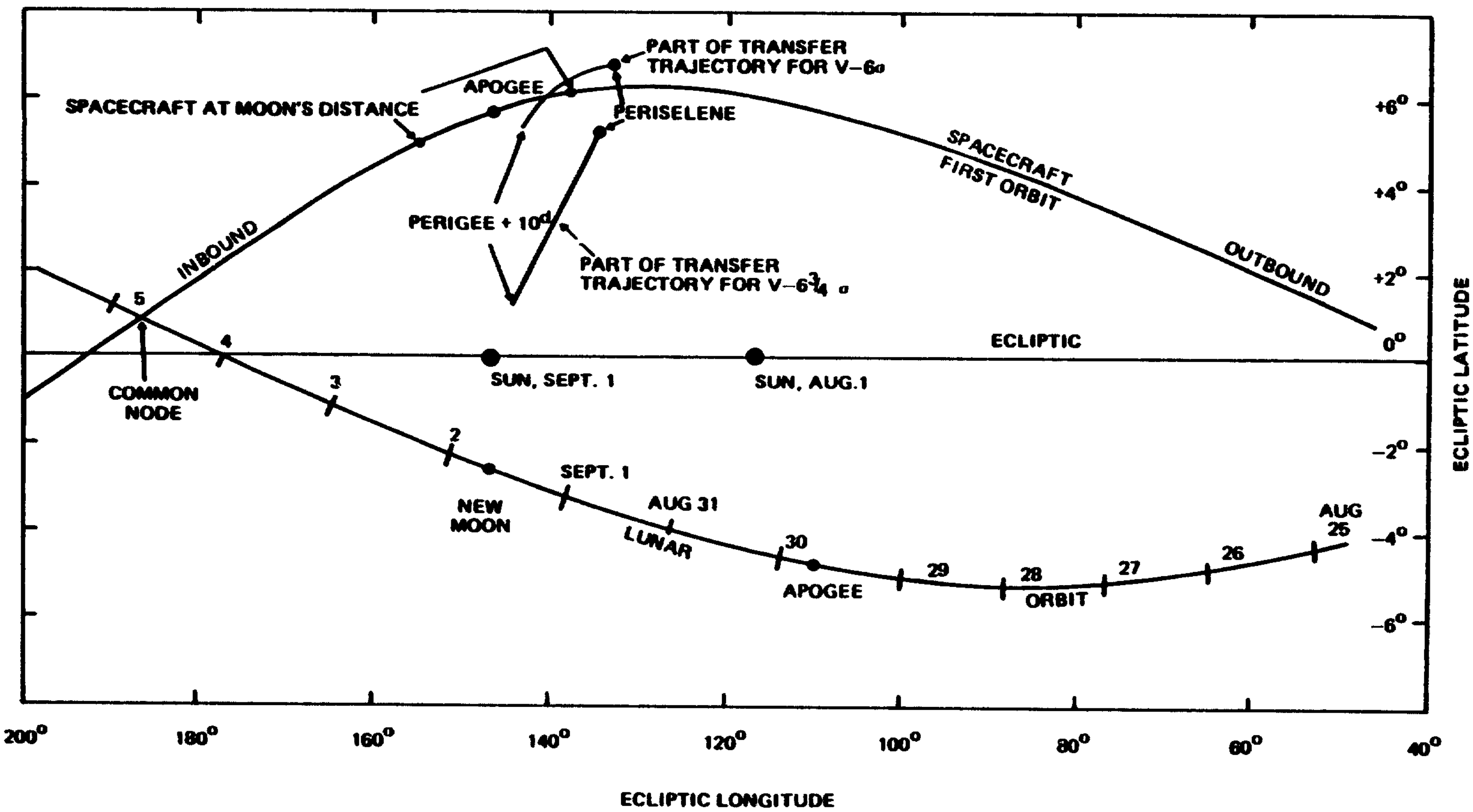

Figure 2-5. Geocentric Zodiacal Plot of the Lunar Orbit and First Orbit of the Spacecraft

thus the results, especially the out-of-plane costs, could differ considerably from those presented in the last several tables. A launch date could be selected where the spacecraft first orbit (or the plane of the originally-planned transfer trajectory) would be nearly coincident with the lunar orbital plane. The lunar out-of-plane perturbations would be insignificant for such a case, and relatively close lunar swingbys might be used to considerable advantage for optimization of the transfer trajectory ΔV costs.

2.5 DELAYED TRANSFERS

The results of Section 2.4 show that there are few contingency cases where the lunar perturbations are so unfavorable that no combination of first orbit and transfer trajectory maneuvers can result in halo orbit insertion for less than the desired 200 meters/second contingency budget. If one is unlucky enough to be in one of these contingency situations where the three-impulse strategy discussed above fails, it might be possible to salvage the mission by letting the spacecraft complete more than one Earth orbit before executing a perigee maneuver for the TTI. The first thought would be to attempt TTI after completing two Earth orbits. There would be considerable motion of the Earth-Sun line away from the spacecraft orbit's line of apsides, as indicated in Figure 2-2. Consequently, the in-plane component of a midcourse correction to achieve a suitable transfer trajectory to the halo orbit would probably be prohibitively large. Two-orbit attempts for the $V - 8\sigma$ case for the August 12 launch resulted in unacceptable ΔV costs of several hundred meters/second.

A delay of about 1 year should be more successful. By then, the apogee of the spacecraft's orbit would again be in the solar direction, and a good transfer trajectory should be achievable with a perigee maneuver. A small maneuver near the first orbit perigee could be executed to change the period of the spacecraft orbit, so that after a year, it would be possible to encounter the Moon for a swingby maneuver to decrease transfer trajectory costs. A two-body model was used to compute approximate costs for these phasing maneuvers listed in

Table 2-18 for the V - 9σ case, July 23rd launch. The number of months and revolutions of the spacecraft (NREV) are given, along with the Julian dates when the Moon and the spacecraft are at the mutual orbital crossing points, with the spacecraft outbound. The difference of these times in days is given under DIFF. Values, in meters/seconds, for ΔVs needed to change the spacecraft period enough to make DIFF vanish are given. The ΔV is assumed to be performed at the apogee or perigee of the first orbit. No value is given in the apogee column if the maneuver lowers perigee, which might result in Earth impact. According to Equation 2-1, the perigee burns are much more efficient. The costs decrease as the number of revolutions increase, allowing the time difference to be divided over a larger number of orbits. The costs become very small after a year. Actual costs for the phasing maneuver were found to differ considerably from values listed in the table since relatively close approaches to the Moon before the desired encounter considerably perturbed the spacecraft orbit, including its period. Figure 2-5 and the discussion of Section 2.4.3.2 show that a lunar encounter would not be very useful for the July and August 1978 launches. A year after launch, the spacecraft orbit would have to have approximately the same orientation as shown in Figure 2-5 in order to keep out-of-plane transfer trajectory maneuver costs within reasonable bounds. A phasing maneuver for a delayed (one year late) transfer could be used to establish perigee at a time of month similar to the favorable August 12 launch V - 6σ transfer, rather than a close encounter with the Moon. A major problem with delayed transfers is a large plane change which the spacecraft orbit usually suffers as a result of lunisolar perturbations during a year. Maintaining the original spacecraft orbit orientation and setting up a good time of month for a low-cost transfer to the halo orbit a year after launch make the calculation of delayed transfers a very difficult problem.

Table 2-18. Delta V Costs for Period Changes for Lunar Encounters for V-9 Sigma, Spacecraft Outbound

MONTH	JD MOON	NREV	JD S/C	DIFF. (days)	APOGEE ΔV (m/sec)	PERIGEE ΔV (m/sec)
1	2443751.28	3	2443754.77	3.488		-7.11
2	2443778.60	5	2443780.28	1.678		-1.67
3	2443805.92	7	2443805.79	-0.132	5.47	0.08
4	2443833.24	9	2443831.30	-1.942	53.46	0.95
5	2443860.57	11	2443856.81	-3.752	78.18	1.47
6	2443887.89	13	2443882.33	-5.562	93.54	1.81
7	2443915.21	16	2443920.59	5.384		-1.50
8	2443942.53	18	2443946.11	3.574		-0.87
9	2443969.85	20	2443971.62	1.764		-0.38
10	2443997.17	22	2443997.13	-0.046	0.60	0.00
11	2444024.50	24	2444022.64	-1.856	20.85	0.32
12	2444051.82	26	2444048.15	-3.666	36.19	0.59
13	2444079.14	28	2444073.66	-5.476	48.32	0.81
14	2444106.46	31	2444111.93	5.470		-0.76

SECTION 3 - LARGE OVERBURNS

There is no strategy for correcting overburn contingency cases equivalent to the three-impulse strategy for underburns described in Section 2. Even a V + 1σ trajectory passes directly through the halo orbit region with the spacecraft soon escaping the Earth and going into a heliocentric orbit. ISEE-3 does not have the ΔV capability for making useful heliocentric orbital changes.

The only real hope for overburns is to perform a retro maneuver as soon as the orbit has been determined, perhaps 18^h after launch, to try to directly establish a transfer trajectory to the halo orbit. The ΔV costs for such a maneuver are similar to those for correcting underburns for direct transfers, like those listed in Table 2-1. Hence, a V + 5σ error could be corrected by a retro maneuver for a total ΔV cost less than the 200 meters/second desired for contingency cases. Somewhat larger overburns could be corrected with higher ΔV costs within the total spacecraft fuel budget, but little fuel would remain for stationkeeping and attitude maneuvers while in the halo orbit, decreasing the spacecraft's useful lifetime.

Heliocentric trajectories were propagated for 1000 days for the V + 3σ, V + 6σ, V + 12σ, and V + 18σ cases to see if there might have been any chance for returning the spacecraft to within a useful communication range of the Earth. For the first three cases, the geocentric distance increased monotonically to to over 10^7 km. The V + 18σ trajectory returned to the Earth-Moon system after 352 days, the hyperbolic perigee distance being 243,000 kms. For this case, a maneuver could be performed to cause a lunar swingby and recapture by the Earth. However, an overburn as large as 18σ at launch was virtually impossible considering the capabilities of the Delta rocket second and third stages used for the transfer trajectory insertion, and thus details of a lunar capture for this case were not investigated.

SECTION 4 - SUMMARY

Details of a contingency strategy developed for the ISEE-3 mission are given in Sections 2 and 3. The purpose of this section is to summarize the results by outlining the procedures which might be followed for large underburn contingency planning for a future libration-point mission.

Before the launch, when a launch date has been selected, direct-transfer costs should be computed for a few contingency cases to allow a table similar to Table 2-1 to be prepared. The results should be similar to those of Tables 2-1 and 2-2. If 1σ is defined to be something other than 5.6 meters/second, the results would be scaled according to the ratio of 1σ to 5.6 meters/second.

The Moon's phase at the selected launch date should be checked. If the Moon is near first quarter (waxing), the contingency costs should be similar to those given in Table 2-10. If the launch is near last quarter (Moon waning), the contingency costs should be similar to those given in Table 2-9.

For constructing a table similar to Table 2-10, a range of contingency cases should be selected. A suggested range of velocity errors would be from 3σ to 10σ, with two or three cases with even larger errors. A suitable interval would be about 5 meters/second (1σ, or 5.6 meter/second, in the case of ISEE-3) in order to obtain enough detail to determine where lunar perturbations may be severe. Tables similar to Table 2-10 should also be prepared for one or two alternate launch dates, especially if the lunar phase is different from the phase at the primary launch date. Contingency cases strongly affected by the Moon should be computed in more detail to determine what first-orbit maneuver might be needed, as discussed in Section 2.4.3 (see Tables 2-16 and 2-17). Any case with real problems which might require a delayed transfer, such as V-8σ for the August 12, 1978, launch, should be determined.

For each of the contingency cases, the orbit should be propagated for one revolution to perigee in order to see if any first-orbit maneuvers are needed, as discussed in Section 2.2. If Earth impact occurs or if perigee is too low, an apogee maneuver is required, as discussed in Section 2.2.2. A small maneuver will probably be needed about 18 hours after launch in order to change the spacecraft's orbital period for adequate tracking station coverage near perigee (see Section 2.2.1). Tabulations similar to those in Tables 2-3 and 2-4 could be useful for estimating an initial ΔV for the detailed calculations which would need to be performed quickly in a real contingency situation. In addition, as soon as the attempted transfer trajectory insertion error is established, the table described above (prepared like Table 2-10) should be consulted to see if a larger maneuver at 18 hours after launch would be needed to avoid large lunar perturbations, as discussed in Section 2.4.3. A transfer trajectory study should quickly be performed for the actual contingency situation to verify the prelaunch results, which should provide a good guide, but will not be highly accurate due to pointing errors. After the first-orbit maneuvers are determined, a study of the new transfer trajectory should be made by varying the perigee ΔV and optimizing the total fuel costs for a midcourse correction performed at apogee. When an optimum perigee ΔV is found in this way, earlier midcourse correction times can be studied. This detailed analysis could be done one or two days after launch rather than before 18 hours after launch, since it is not critical (see Section 2.4.2).

If the actual contingency case poses relatively severe problems with lunar perturbations, such as the V-8σ case for the 1978 August 12 launch, some delayed transfers might be computed, as described in Section 2.5. But since such a study would probably not yield a useful result within a day after launch, the optimum single-orbit case would probably have to be selected. A detailed delayed transfer study could be made during the next several days to determine if there might be a delayed transfer which would have lower fuel costs than the selected single-orbit case.

Details of the perigee maneuver can be computed during the first orbit. Section 2.3 gives an approximate guide for this calculation. While performing the contingency transfer trajectory calculations using an impulsive ΔV at perigee, the analyst should keep in mind that the actual ΔV required in terms of fuel costs will likely be 20 percent or more greater than the perigee impulse. This can be inferred from Table 2-7, assuming that axial jets are used with the burn started at penumbral exit.

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