NASA CR-137524

STUDY OF BALLISTIC MODE COMET ENCKE MISSION OPPORTUNITIES

FINAL REPORT

August 1974

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Prepared Under Contract No. NAS2-7564

by

MARTIN MARIETTA CORPORATION
Denver, Colorado 80201

for

AMES RESEARCH CENTER
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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MCR 74-210

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by

G. R. Hollenbeck & J. M. Van Pelt

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Approved: G. R. Hook

G. R. Hook
Program Manager

FOREWORD

This report documents recent study effort conducted under Contract NAS2-7564 for Ames Research Center under the direction of the NASA Contract Monitor E. L. Tindle.

A prior phase of the contract was concerned with mission and spacecraft feasibility for a 1980 fast flyby mission to comet Encke. Results are documented in a summary report (NASA CR-114670) and a technical report (NASA CR-114671).

This phase of the effort was addressed to performance and navigation analysis of the advanced Encke mission modes of slow flyby and rendezvous. Material is also presented relative to the prospects of a sample return mission in the era of Space Shuttle availability.

ACKNOWLEDGEMENTS

The following individuals contributed significantly to the conduct of this study and preparation of this report. Their efforts in the indicated disciplines have enhanced the quality of the data presented and are greatly appreciated.

P. C. Rockenbach - Computer Programming and Mission Analysis

R. J. Boain - Encke Ephemeris Error Analysis

D. V. Byrnes - Navigation Analysis

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I. SUMMARY

I. SUMMARY

Study contract NAS2-7564 has been concerned with ballistic mode missions to the short period Comet Encke. In particular, the feasibility of comet exploration based on Pioneer-class spacecraft in combination with programmed launch vehicles and conventional spacecraft propulsion has been addressed.

A prior phase of this study was devoted to a unique mission opportunity corresponding to 1980 launch and characterized by encounter geometry producing flythrough of the comet coma and tail with a relative velocity of about 18 km/sec. Options for extending the mission to include flyby of the asteroids Geographos and Toro were also assessed. Emphasis was placed on design feasibility of the spacecraft (and associated coma probe) and evaluation of the expected science data return. Documentation of this study phase is presented in a summary report (NASA CR-114670) and a technical report (NASA CR-114671).

Figure I-1 depicts the short flight time 1980 fast flyby mission with encounter occurring about 16 days before comet perihelion. This flight profile is compatible with the Atlas/Centaur/TE364-4 launch vehicle. Options exist (with higher launch energies) to accomplish encounter near the comet perihelion at relative velocities as low as about 7 km/sec. These missions, requiring a larger launch vehicle, are also shown on Figure I-1. Use of the Titan IIIE/Centaur/TE364-4 combination would provide sufficient performance to accommodate a Mariner-class spacecraft or alternatively, propulsion to reduce relative velocity to about 5 km/sec for a Pioneer-class spacecraft. This latter case was considered in the current study phase as a candidate for the slow flyby class of Encke mission options.

The advanced mission modes of slow flyby and rendezvous generally require launch from the vicinity of the comet descending node and involve relatively long flight times. As illustrated on Figure I-2, large midcourse maneuvers are required to achieve encounter in the region of perihelion necessitating use of a large launch vehicle such as Titan IIIE/Centaur. Several opportunities for advanced Encke missions occur during the 1980 time frame. In general, flight times of about 2 years represent slow flyby missions within the constraints of conventional chemical spacecraft propulsion. For the longer flights depicted on Figure I-2 (on the order of 3 years), rendezvous missions are feasible.

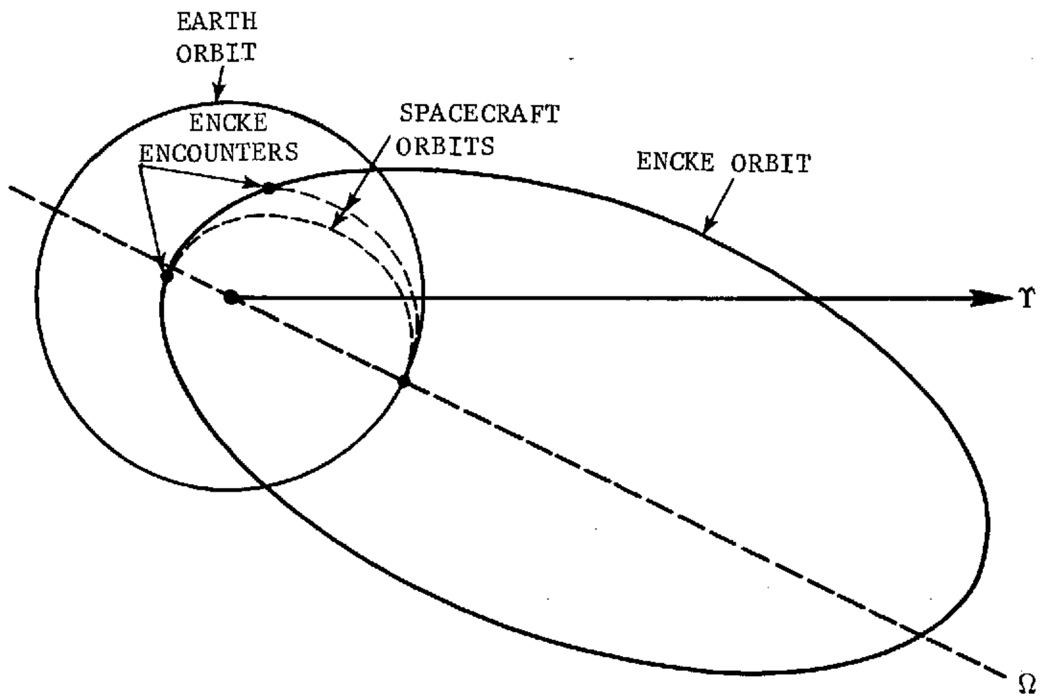

FIGURE I-1 TYPICAL FLIGHT PROFILES FOR ENCKE FAST FLYBY MISSIONS

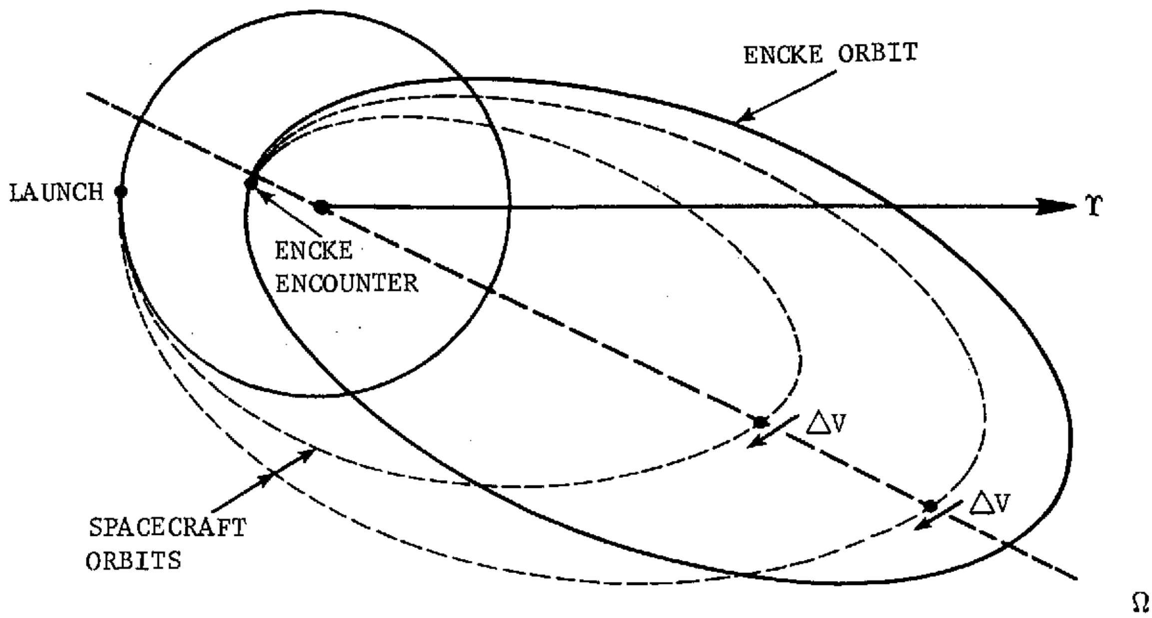

FIGURE I-2 TYPICAL FLIGHT PROFILES FOR ENCKE SLOW FLYBY AND RENDEZVOUS MISSIONS

The study contract phase reported in this document involved complete parametric performance analyses for the ascending node launch case and 7 selected cases of launch from the vicinity of the descending node. In addition, time histories, encounter geometries and navigation requirements were analyzed for representative flight profiles. Encke ephemeris errors were assessed and, while large in comparison to spacecraft dispersions, do not appear to jeopardize mission feasibility.

A summary of performance potential for the missions studied is presented in Table I-1. These data are predicated on the Titan IIIE/Centaur launch vehicle, chemical spacecraft propulsion and other qualifying conditions as specified.

Five missions of the slow flyby type are identified on Table I-1 with flight time varying from about 4 months to about 2½ years. The indicated trend of longer flight time for slower flyby velocity is due to the nature of the Encke orbit geometry. Also, the longer flight times generally correspond to greater performance capabilities. Timely launch opportunities are available to support mission planning options such as backup or follow-on to the 1980 fast flyby mission currently under consideration by NASA and/or to provide precursory experience for the more demanding rendezvous mission.

Best performance for Encke rendezvous missions generally corresponds to encounter in the region of comet perihelion. Table I-1 presents capabilities for three such opportunities.

Under certain conditions of Earth-Encke orbit phasing, rendezvous missions can be achieved with encounter as early as the region of comet aphelion. For example, flight time for the 1984 launch opportunity can be reduced by 16 months (from a nominal value of 40 months) for a performance loss of about 25%. In the case of 1987 launch, no significant performance penalties are involved for flight time reductions as great as 2 years. Geometry misalignments are somewhat worse for 1990 launch (not analyzed for perihelion rendezvous) but aphelion-class rendezvous has been confirmed as a usable option. The three cases included in Table I-1 for early rendezvous represent the only such opportunities in the time period of interest.

Projected availability of the Shuttle/Centaur launch vehicle or equivalent would provide spacecraft weight capabilities of about 500 kg and 400 kg for the 1987 and 1990 missions respectively. These opportunities could be utilized to launch Mariner-class rendezvous spacecraft or, alternatively, a modest sample return mission may be possible with Pioneer technology. Performance requirements for the Earth return phase have been evaluated and are compatible with comet stopover intervals of a few months. Total trip time from launch through sample return would be about 44 months which provides an attractive option for investigators interested in this ultimate of comet science objectives.

TABLE I-1 PERFORMANCE POTENTIAL OF ENCKE MISSION OPTIONS

MISSION TYPE	EARTH LAUNCH	ENCKE ENCOUNTER	FLIGHT TIME (MONTHS)	NET S/C WT. AT ENCOUNTER (KG)
2.3 KM/SEC Flyby	3-79	12-80	21.2	245
* 5.0 KM/SEC Flyby	8-80	12-80	3.6	295
1.7 KM/SEC Flyby	3-82	4-84	24.9	285
1.0 KM/SEC Flyby	3-85	7-87	28.2	290
0.7 KM/SEC Flyby	3-88	10-90	31.6	310
Perihelion Rendezvous	3-81	3-84	36.5	295
Perihelion Rendezvous	3-84	7-87	40.2	320
Perihelion Rendezvous	3-87	10-90	43.6	340
Aphelion-Class Rendezvous	3-84	3-86	24.0	240
Aphelion-Class Rendezvous	3-87	10-88	19.7	340
Aphelion-Class Rendezvous	2-90	5-92	26.7	270

Conditions: Launch from vicinity of Encke descending node (except as noted)

Launch Period = 10 days

Launch Vehicle = Titan IIIE/Centaur + TE364-4 as required

Midcourse correction allowance = 150 mps

Spacecraft Propulsion (except as noted)

Two-Stage, Earth-Storable Bi-Propellants (equal propellant load for each stage)

Specific Impulse = 310 sec, Propellant Fraction = 0.9

* Launch from Encke ascending node, single-stage spacecraft propulsion.

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II. INTRODUCTION

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Prior study effort under contract NAS2-7564 was concerned with Encke missions characterized by fast flyby (15-25 km/sec) and compatibility with Atlas/Centaur TE364-4 launch and Pioneer-class spacecraft. The study phase documented in this report extends consideration of ballistic flight capabilities to the more difficult Encke mission modes of slow flyby (defined by science considerations as 5 km/sec and less) and rendezvous. For these latter mission categories, Titan IIIE/Centaur launch vehicles are required to provide adequate performance for Pioneer-class spacecraft.

Two basic types of flight geometry were considered. Figure II-1 depicts a single case of ascending node launch which applies to 1980 launch and the 1980 apparition of Encke. Encounter near Encke perihelion corresponds to minimum achievable flyby velocity of about 7 km/sec. With Titan IIIE/Centaur/TE364-4 launch capabilities, spacecraft propulsion can be provided to reduce relative velocity to 5 km/sec at Encke encounter. This flight option does not repeat for 33 years and was included in the study scope to complete the perspective of Encke mission opportunities in the time period of interest.

The remainder of the mission cases studied are of the type represented on Figure II-2. As shown, these missions are characterized by launch from the vicinity of the Encke descending node, large midcourse maneuvers near the ascending node to simultaneously lower perihelion and deflect the spacecraft into the Encke orbit plane, and encounter near Encke perihelion to minimize performance requirements. As indicated on the figure, a wide range of transfer orbits is possible with corresponding variations in flight time and performance parameters.

To provide a basis for selection of specific missions warranting detailed investigation, simplified analytical data were generated. Figure II-3 displays the primary interactions between time-of-flight and performance characteristics. These data are quantitatively imprecise but provide valid indications of important trends.

The common technique of summing velocity maneuvers (including launch from Earth park orbit) is employed to represent performance requirements. As shown, higher launch energies produce longer flight times by increasing the periods of the spacecraft transfer orbits. A compensating effect is

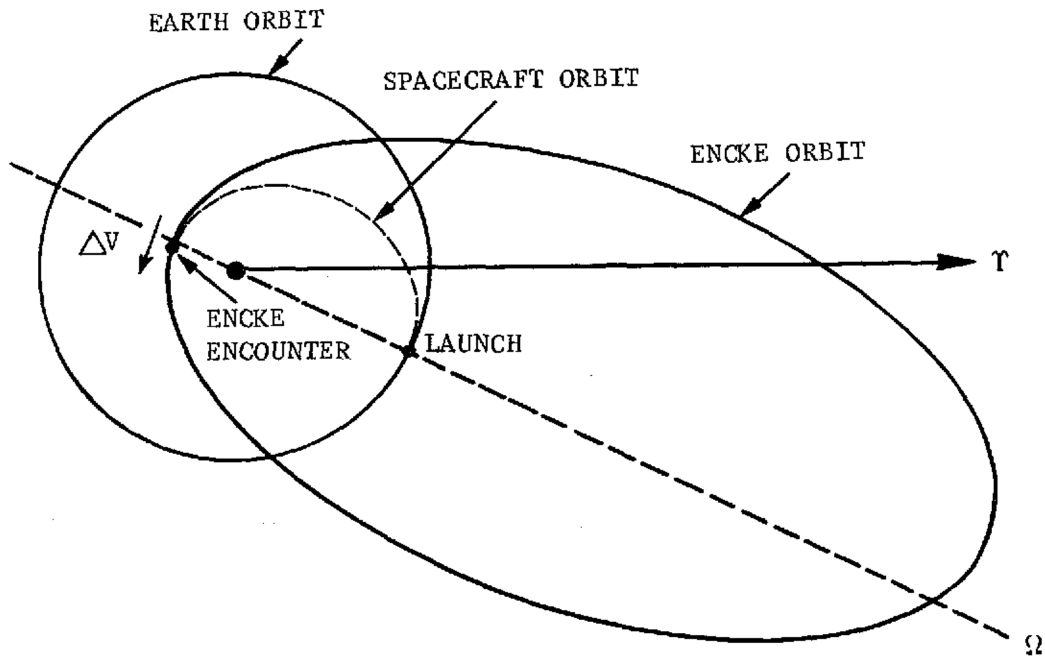

FIGURE II-1 HELIOCENTRIC GEOMETRY FOR ASCENDING NODE LAUNCH

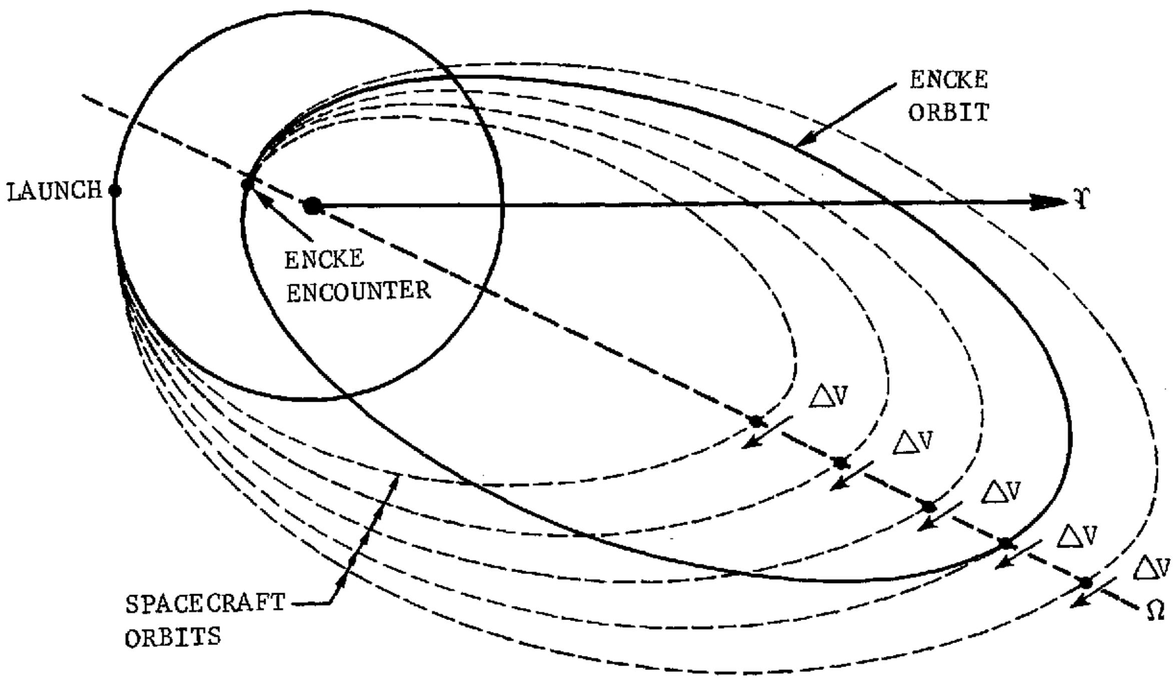

FIGURE II-2 HELIOCENTRIC GEOMETRIES FOR DESCENDING NODE LAUNCH reduction of the orbit-change velocity maneuver when executed at larger heliocentric radius. The sum of these two velocity increments is shown to be relatively insensitive to time-of-flight.

The relative velocity at Encke encounter is more dependent on flight geometry. For flyby missions, this parameter characterizes the science instrument requirements and potential science return but does not reflect directly on performance. However, for rendezvous missions, the encounter velocity must be arrested with spacecraft propulsion and represents a primary performance consideration.

For short flight times, the spacecraft orbit is constrained inside the Encke orbit, and the final velocity maneuver is posigrade to match the comet velocity. For longer flight times corresponding to flight outside the Encke orbit, the final maneuver must be retrograde. A special case, involving about 3.7 years between launch and Encke perihelion passage, exhibits a zero value for the third velocity maneuver. As shown by Figure II-3, total velocity requirements are minimized for this condition, producing a theoretical optimum flight time for Encke rendezvous missions, and illustrate the disadvantages of excessive flight time.

The data of Figure II-3 were generated from parametric variation of the heliocentric radius at which the large midcourse velocity maneuver was applied. To interpret these data in terms of actual launch opportunities in the time period of interest (through the 1980's), the launch to encounter intervals for actual flight geometries are indicated at the top of the Figure. The Encke apparitions of 1980, 1984, 1987 and 1990 are considered in combination with the yearly passages of Earth through the Encke descending node. As shown, several cases are encompassed by the time-of-flight range, and the relative performance potential of each case can be judged for either the slow flyby or rendezvous mission modes.

Not all of the cases depicted on Figure II-3 were subjected to detailed analysis since the analytical trends provide a valid basis for inferring the performance potential of intermediate cases. Also, considerations of launch date feasibility and excessive flight times were employed to constrain the study scope. The descending node launch cases selected for further investigation are specified on Figure II-3 in terms of launch/encounter date.

CONDITIONS: LAUNCH IN ECLIPTIC FROM ENCKE DESCENDING NODE

CO-PLANAR ENCOUNTER AT ENCKE PERIHELION

ENCKE LINE OF APSIDES ASSUMED COINCIDENT WITH ENCKE LINE OF NODES

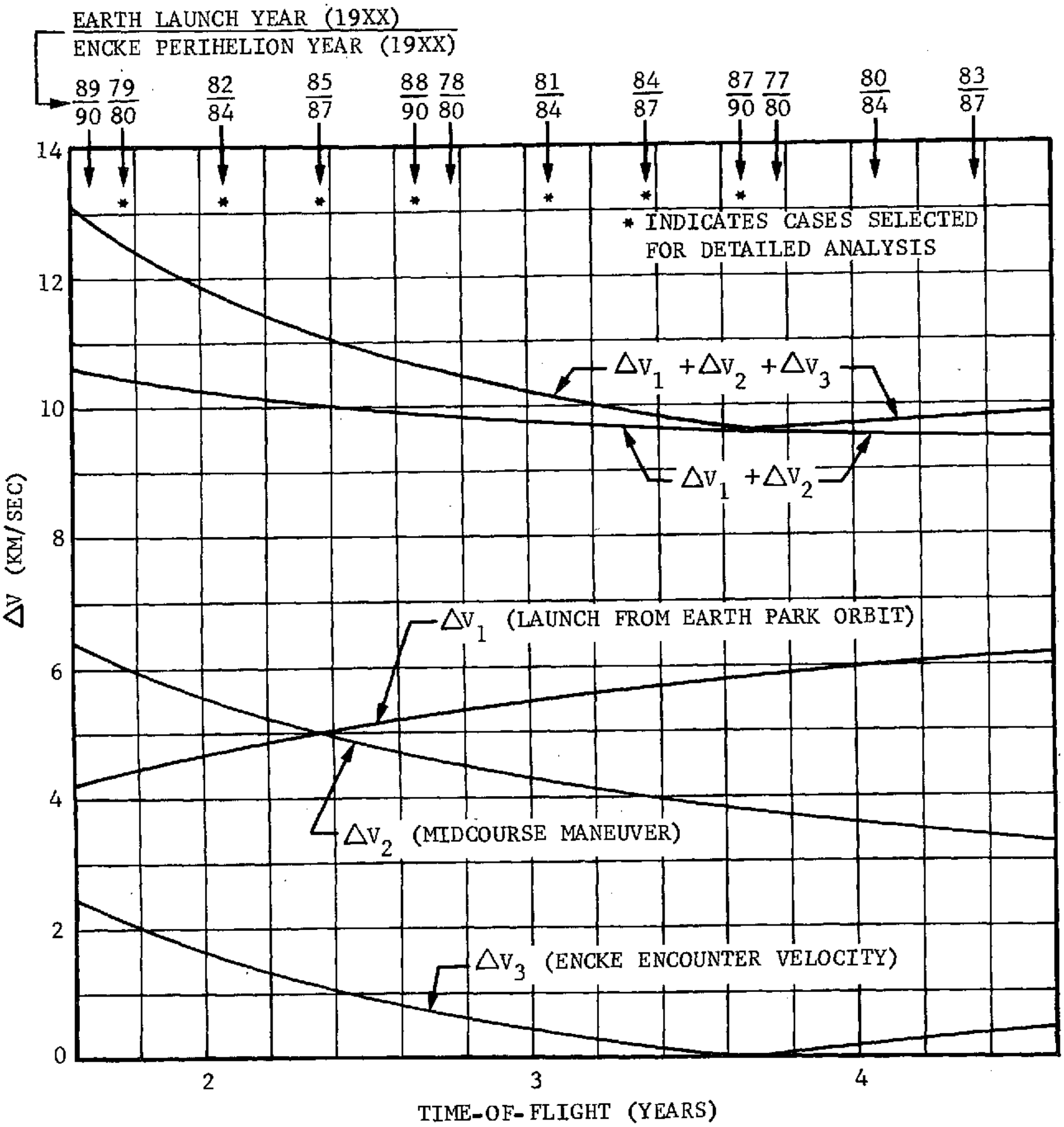

FIGURE II-3 TIME-OF-FLIGHT EFFECTS FOR DESCENDING NODE LAUNCH

The levels of treatment accorded each mission opportunity are summarized in Table II-1. Performance characteristics were defined for each case while more detailed investigations, such as assessment of navigation requirements, were limited to a few representative mission profiles. Results of the specified study tasks are presented in Sections III through V of this document.

TABLE II-1 SUMMARY OF STUDY SCOPE

MISSION CASES		STUDY TASKS			
LAUNCH YEAR	#ENCOUNTER YEAR	PERFORMANCE ANALYSIS	TRAJECTORY HISTORIES	ENCOUNTER GEOMETRIES	NAVIGATION ANALYSIS
1979	1980	X			
*1980	1980	X	X	X	
1981	1984	X			
1982	1984	X	X	X	X
1984	1987	X	X	X	X
1985	1987	X			
1987	1990	X			
1988	1990	X			

All cases encounter Encke in the vicinity of Encke perihelion.

* Launch from vicinity of Encke ascending node (all other cases launch from vicinity of Encke descending node).

X Included in study scope.

Preliminary data have been generated for two subjects not covered by Table II-1. Evaluations of Encke ephemeris errors and options for rendezvous in the vicinity of Encke aphelion are documented in Sections VI and VII respectively.

III. PERFORMANCE CHARACTERISTICS

III. PERFORMANCE CHARACTERISTICS

This section is concerned with the performance characteristics for all of the missions where spacecraft arrival occurs near Encke perihelion. (Aphelion-class encounters are discussed in Section VII.) A description of the analysis is given, including the method of trajectory optimization and the selection of the presentation format. Both graphical and tabular results are presented for the various mission opportunities, with final tabular data summarizing performance characteristics and presenting representative payload potential with applicable launch vehicles.

A. ANALYSIS DESCRIPTION

1. 1980 Ascending Node Launch

The current phase of this contract is primarily concerned with the longer missions (2-3 years) which are launched from the descending node (as shown in Figure II-2) and require a maneuver near the spacecraft-orbit aphelion. An interesting opportunity also exists to achieve a slow flyby (approach velocity of 5 km/sec or less) by launching when the Earth is near the comet ascending node (as shown in Figure II-1) and arriving at the comet within 4 months. The latter mission does not require an intermediate maneuver and is discussed under Section III.B.

2. Descending Node Launches

The descending node launches covered under this contract phase involve Comet Encke encounters for the 1980, 1984, 1987, and 1990 passages. Comet ephemeris data for 1980 were identical to that used in the earlier phase of this contract and for the previously mentioned ascending node launch. Comet ephemeris data for the 1984, 1987, and 1990 passages were generated with an n-body trajectory program initiated at the 1980 perihelion and integrated through the 1990 perihelion point (see Appendix A).

The data range of interest for the parametric performance analysis covers a 30-day launch interval from late February (just after the Earth passes through the comet descending node) until late March and an 80-day comet encounter period, centered on the comet perihelion, for all years investigated.

In all cases, a velocity maneuver is required near the spacecraft orbit aphelion. This maneuver varies from 4 to 6 km/sec, depending upon the mission. For flyby missions, this is the only maneuver required. For rendezvous mission opportunities, an additional maneuver of from less than one hundred to several hundred meters per second is required.

ΔV_{TOT}, which is the sum of ΔV_1 (velocity increment required to leave low Earth orbit for the required launch energy), ΔV_{MC} (the mid-course velocity maneuver), and V_H (the approach velocity at encounter), has been minimized. This optimization procedure has taken place for all trajectories presented and is a good indicator of maximum performance for both the flyby and rendezvous missions. (Leaving out V_H in the minimizing procedure for flyby trajectories permitted the approach velocity to become unreasonably large for a small reduction in the sum of ΔV_1 and ΔV_{MC}, and was therefore not used).

In the optimization procedure, two simple conic legs are flown (from launch to the midcourse maneuver, and from the midcourse maneuver until comet encounter). The parameters involved are launch C_3 and the orbit inclination of the first leg, and the position (and corresponding time) and the magnitude and direction of ΔV_{MC} (which determines the orbit inclination of the second leg). This procedure was carried out for all of the individual trajectories.

It has been observed in some of the missions that launch declinations exist which require launches slightly off of due East at some extremities of the launch/arrival grid. The ΔV penalties observed do not normally exceed 50 m/sec and have, therefore, been treated as a second order effect upon performance and not significant to the purposes of this study.

During the early portion of this contract phase, an attempt was made to use a presentation technique similar to the ascending node cases, namely a contour type of presentation showing the significant performance parameters over the region of interest. It was found that although the ΔV_{TOT} parameter was a well behaved function, the individual components, particularly C_3 and ΔV_{MC}, were complex and of limited usefulness in interpreting the data. An example of the contour method of presentation (1979/80 opportunity) is shown in Appendix B. The selected presentation method is discussed in Section III.B.

B. PERFORMANCE RESULTS

1. 1980 Ascending Node Launch

Complete 1980 Encke parametric performance data was generated in the previous phase of this contract for the region of spacecraft arrival 32 days to 6 days before comet perihelion passage. These missions were all fast-flyby encounters with approach velocities varying from 12 km/sec to 28 km/sec over the full data range.

Since the completion of that phase of the work, interest has shifted to the near-perihelion encounter region where the possibility of a slow flyby (with a final maneuver) exists. Figure III-1 presents the results of a study which concentrates upon the region of Encke encounters from 6 days before perihelion to 8 days after perihelion and launches within 10 days of Earth passage through the comet ascending node.

The right portion of Figure III-1 shows the velocity contours reaching a minimum value of 7.05 km/sec for encounters about 2 days after perihelion passage (Type II trajectories). By superimposing the 7.5 km/sec and 8 km/sec contours on the left side of the figure, a 10-day launch period can be selected. The best single comet encounter date for minimizing both C_3 and V_H is 5.5 days after perihelion. Maximum C_3 required is 99.1 km^2/sec^2 and maximum V_H is 7.88 km/sec during the 10-day interval. A velocity maneuver of 2.88 km/sec would be required near encounter to reduce the flyby velocity to 5 km/sec. Table III-1 shows a typical trajectory printout (see Appendix C for printout key) for a launch in the middle of the 10-day launch period (8-23-80) and an arrival at 5.5 days after perihelion (12-12-80). Note that the C_3 and V_H values are lower than maximum at this time. A further discussion of launch date effects is given in Section IV.

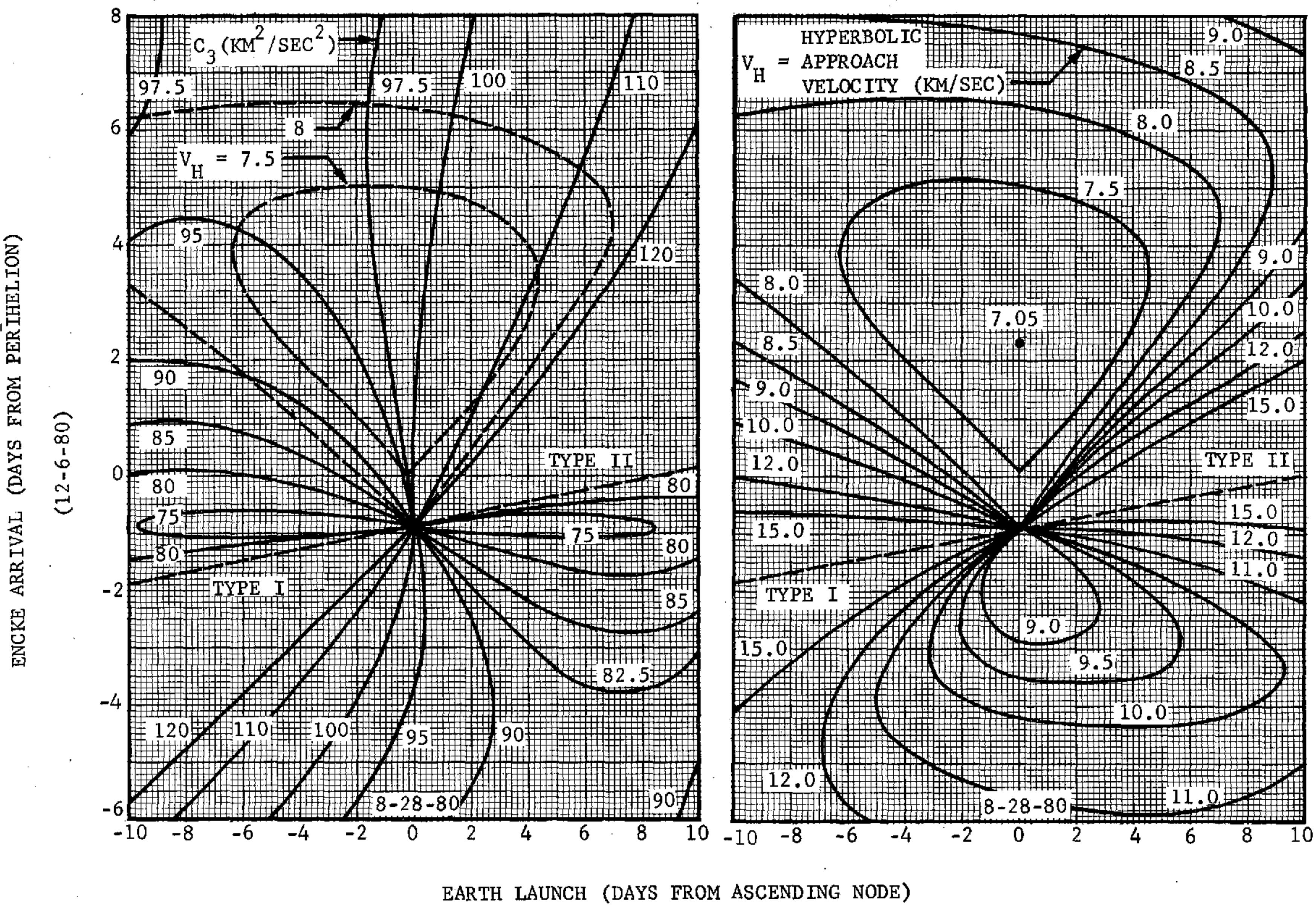

FIGURE III-1 PERFORMANCE PARAMETERS FOR 1980 LAUNCH/1980 ENCOUNTER

TABLE III-1 TRAJECTORY PRINTOUT FOR TYPICAL 1980 LAUNCH/1980 ENCOUNTER

LAUNCH			
LAUNCH DATE	2444474.625 -	8/23/1980. 3. 0. 0	
C3	96.150		
DLA	2.221		
L-TARG. ANGLE	225.589		
DV1	6.961		
PERIOD	201.951		
PERIHELION	.336		
APHELION	1.012		
STATE	.13015857E+09	-.77067045E+08	.44637614E+04
	.10974399E+02	.17368505E+02	.39639673E+01
ELEMENTS	.10077596E+09	.50158855E+00	.10924844E+02
	-.30638585E+02	-.17837261E+03	.17838153E+03
SPHERICAL	.15126329E+09	.16907915E-02	.32937017E+03
	.20924040E+02	.10920432E+02	.57712998E+02

TARGET			
ARRIVAL DATE	2444585.542 -	12/12/1980. 1. 0.28	
VHP	7.671		
DVSUM	14.633		
PHASE	117.928		
SEV	16.437		
(-S) VE	-131.235		
RANGE	1.188		
STATE	-.53173504E+08	-.13687242E+08	-.75036396E+07
	.10884551E+01	-.58140663E+02	-.95480215E+01
ELEMENTS	.10077596E+09	.50158855E+00	.10924844E+02
	-.30638585E+02	-.17837261E+03	.17838153E+03
SPHERICAL	.55417206E+08	-.77819069E+01	.19443498E+03
	.58929501E+02	-.93244236E+01	.27107251E+03

2. Descending Node Launches

Figures III-2 through III-8 are examples of the method chosen for data presentation. This method was found to be more convenient for interpreting the data, since all parameters of interest can readily be compared at readable scales.

In addition to the figures, which completely summarize the parametric performance for the various mission opportunities, a typical trajectory printout has been presented for each case (See Appendix C for printout key). No attempt has been made to indicate best performance in these figures, since science requirements will probably be mission-peculiar with respect to arrival time. The sample trajectory printouts use arrival at perihelion passage in all cases with launches corresponding to the center of a high-performance 10-day launch period.

All figures show the important parameters of C_3, midcourse velocity maneuver, and hyperbolic approach velocity for a 30-day post-nodal period of launches and an 80 day comet-encounter period, centered on perihelion. In all cases, the conic trajectory legs from launch to the midcourse velocity maneuver are Type I trajectories. The second conic leg (from maneuver to Encke encounter) is Type I for encounters up to about 1 day before perihelion and Type II for encounters after this region.

The trajectory printouts, in addition to the parameters shown on the figures include launch declination, heliocentric transfer angles, orbital characteristics of the individual trajectory legs, and spacecraft states at launch, after the maneuver, and at comet encounter. Also presented are approach phase angles at encounter and spacecraft/Earth range and angles relative to Earth at the time of encounter.

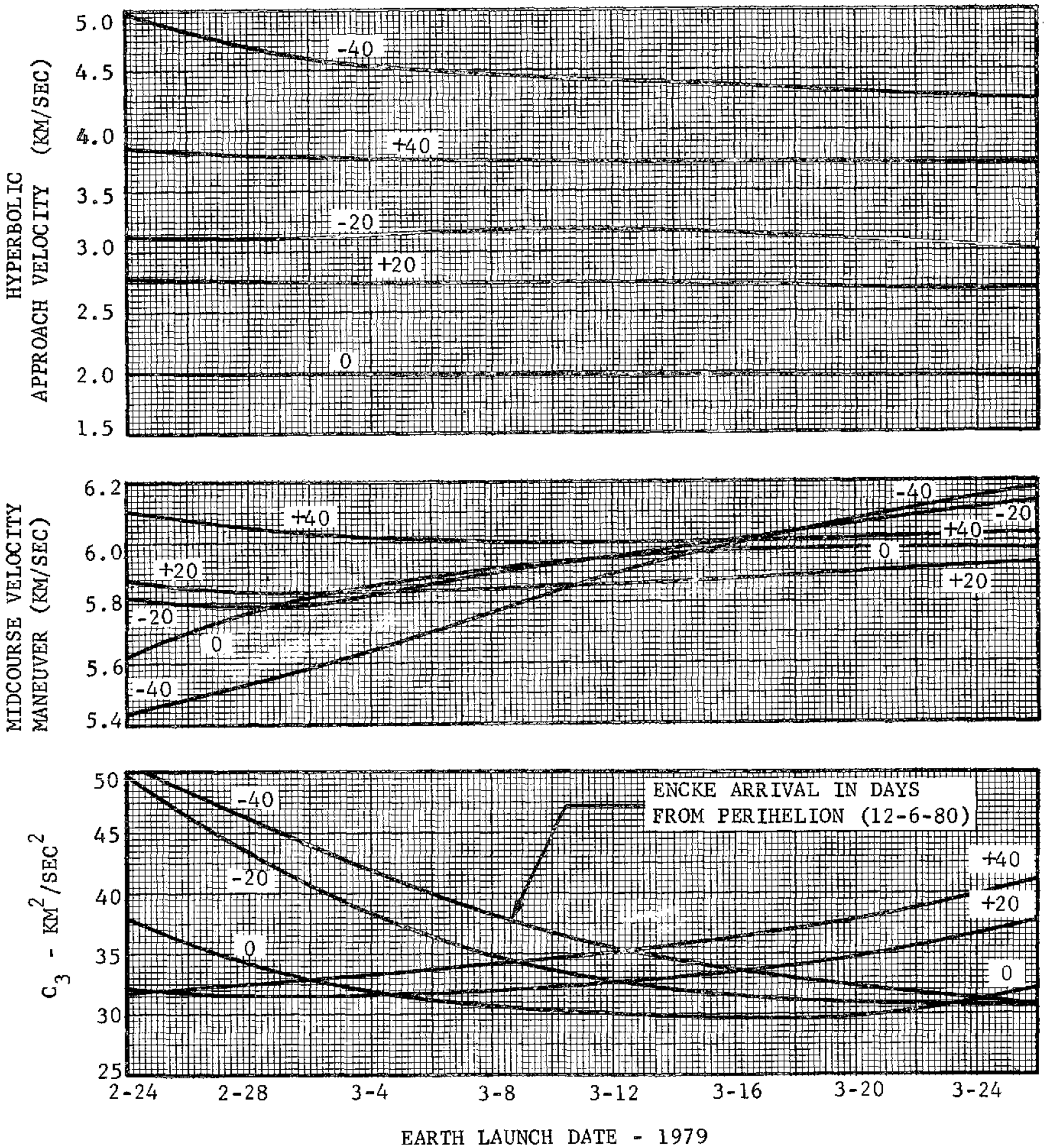

FIGURE III-2 PERFORMANCE PARAMETERS FOR 1979 LAUNCH/1980 ENCOUNTER

TABLE III-2 TRAJECTORY PRINTOUT FOR TYPICAL 1979 LAUNCH/1980 ENCOUNTER

LAUNCH			
LAUNCH DATE	2443934.500 -	3/ 2/1979. 0. 0. 0	
C3	33.127		
DLA	-33.485		
L-MAN. ANGLE	173.175		
DV1	4.653		
PERIOD	776.573		
PERIHELION	.988		
APHELION	2.319		
STATE	-.13937951E+09	.50515297E+08	-.24301988E+04
	-.10584867E+02	-.33775666E+02	-.10017247E+01
ELEMENTS	.24735161E+09	.40266686E+00	.16227060E+01
	-.19955182E+02	-.17117577E+03	-.87910629E+01
SPHERICAL	.14825129E+09	-.93921704E-03	.16007797E+03
	.35409582E+02	-.16210939E+01	.25259967E+03
MANEUVER			
MANEUVER DATE	2444258.969 -	1/20/1980.11.15.21	
DELTA V	5.827		
DVSUM	10.479		
MAN.-T ANGLE	186.994		
PERIOD	554.727		
PERIHELION	.340		
APHELION	2.303		
STATE	.30231360E+09	-.15233804E+09	-.11336343E+07
	.72914953E+01	.74266379E+01	.20752550E+01
ELEMENTS	.19765734E+09	.74272710E+00	.11883705E+02
	-.25832068E+02	-.17461463E+03	.17368287E+03
SPHERICAL	.33852869E+09	-.19186729E+00	.33325613E+03
	.10612612E+02	.11276628E+02	.45526078E+02
TARGET			
ARRIVAL DATE	2444580.079 -	12/ 6/1980.13.53.45	
VHP	2.010		
DVSUM	12.490		
PHASE	81.118		
SEV	19.404		
(-S) VE	-105.667		
RANGE	1.021		
STATE	-.47805799E+08	.17304614E+08	-.11058845E+07
	-.22503474E+02	-.62053806E+02	-.13816942E+02
ELEMENTS	.19765734E+09	.74272710E+00	.11883705E+02
	-.25832068E+02	-.17461463E+03	.67658111E+00
SPHERICAL	.50853388E+08	-.12460823E+01	.16010089E+03
	.67438780E+02	-.11822548E+02	.25006710E+03

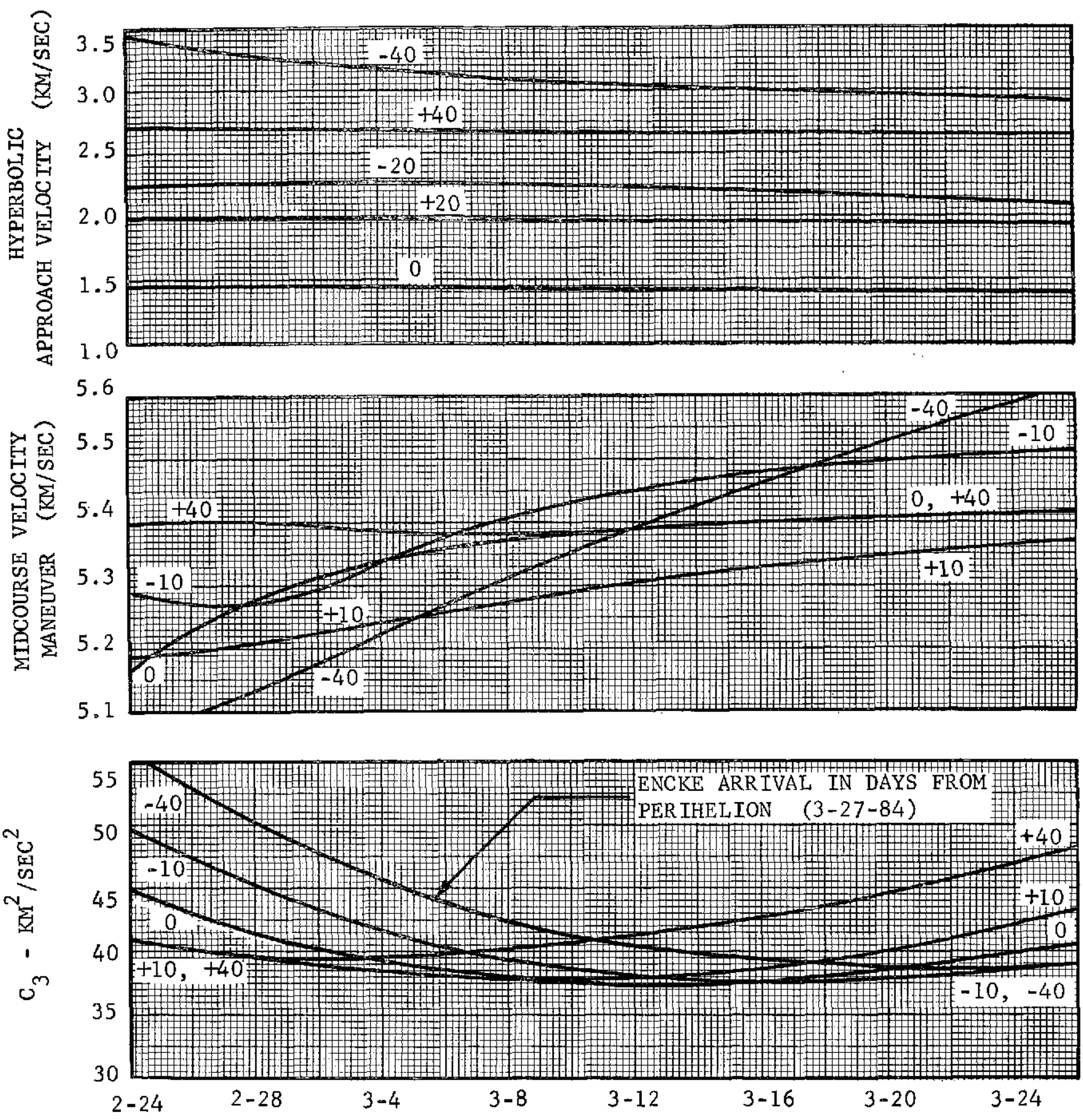

FIGURE III-3 PERFORMANCE PARAMETERS FOR 1982 LAUNCH/1984 ENCOUNTER

TABLE III-3 TRAJECTORY PRINTOUT FOR TYPICAL 1982 LAUNCH/1984 ENCOUNTER

```
LAUNCH
   LAUNCH DATE    2445030.500 -   3/ 2/1982. 0. 0. 0
   C3                  40.343
   DLA                -31.364
   L-MAN. ANGLE       173.098
   DV1                  4.938
   PERIOD             894.304
   PERIHELION            .988
   APHELION             2.645
   STATE          -.13956072E+09   .50037334E+08   -.26426381E+04
                  -.10757236E+02  -.34430289E+02   -.87412326E+00
   ELEMENTS        .27175862E+09   .45593872E+00    .13894108E+01
                  -.19766640E+02  -.17237386E+03   -.75840212E+01
   SPHERICAL       .14825948E+09  -.10212636E-02    .16027547E+03
                   .76082217E+02  -.13881762E+01    .25264940E+03

MANEUVER
   MANEUVER DATE  2445399.709 -   3/ 6/1983. 5. 0.57
   DELTA V              5.315
   DVSUM               10.253
   MAN.-T ANGLE       186.920
   PERIOD             661.205
   PERIHELION            .341
   APHELION             2.630
   STATE           .34452677E+09  -.17271332E+09   -.11162222E+07
                   .68760742E+01   .63863000E+01    .18309518E+01
   ELEMENTS        .22220329E+09   .77042766E+00    .11882360E+02
                  -.25836179E+02  -.17443031E+03    .17362475E+03
   SPHERICAL       .38539568E+09  -.16594610E+00    .33337512E+03
                   .95629831E+01   .11093085E+02    .42985051E+02

TARGET
   ARRIVAL DATE   2445787.169 -   3/27/1984.16. 3.21
   VHP                  1.459
   DVSUM               11.712
   PHASE               80.966
   SEV                -12.306
   (-S)VE              38.598
   RANGE                 .709
   STATE          -.47339887E+08   .17401751E+08   -.11004189E+07
                  -.22598786E+02  -.62465363E+02   -.13901910E+02
   ELEMENTS        .22220329E+09   .77042766E+00    .11882360E+02
                  -.25836179E+02  -.17443031E+03    .44399181E+00
   SPHERICAL       .51012397E+08  -.12360574E+01    .16004955E+03
                   .67866706E+02  -.11820214E+02    .25011083E+03
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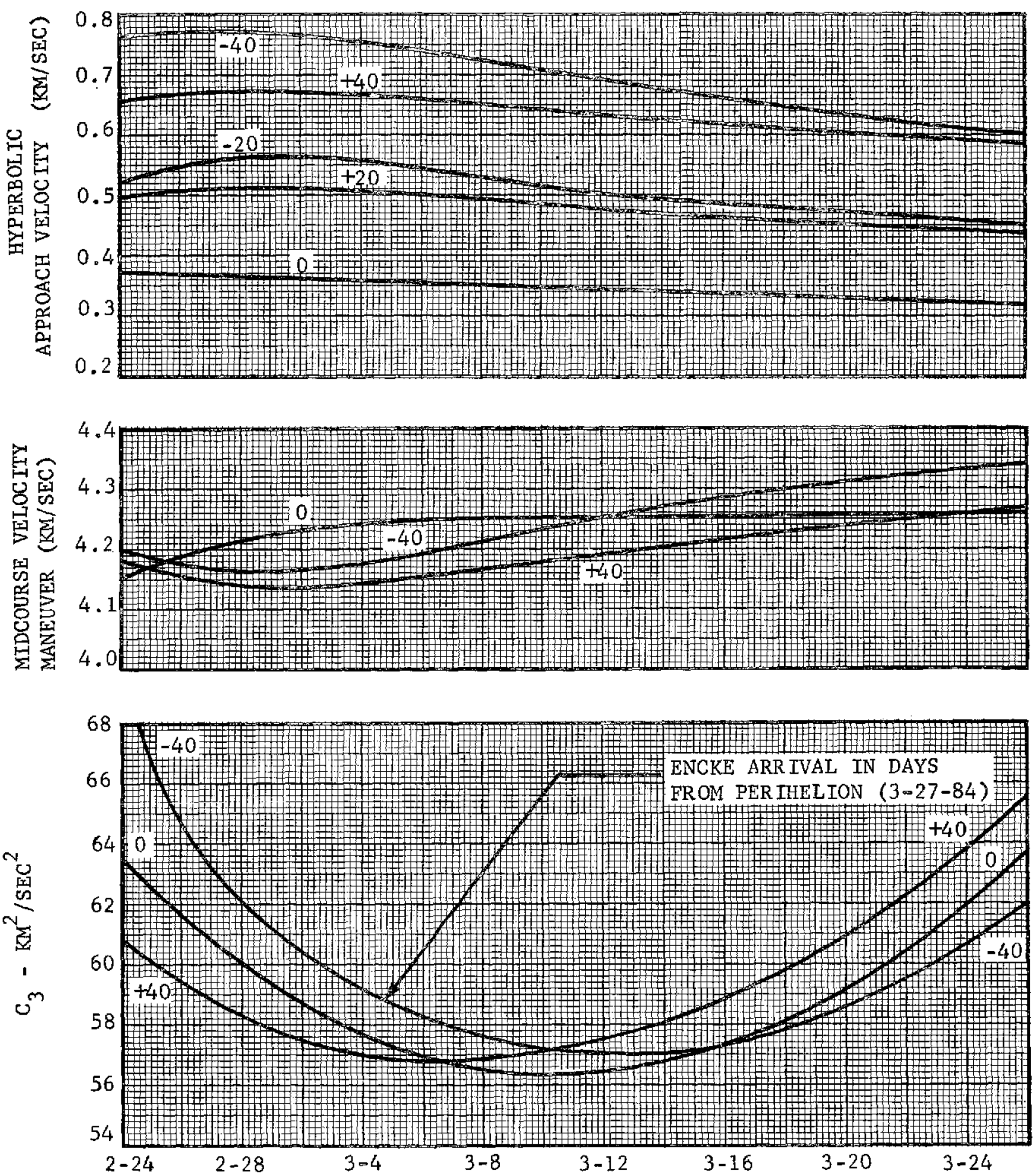

FIGURE III-4 PERFORMANCE PARAMETERS FOR 1981 LAUNCH/1984 ENCOUNTER

TABLE III-4 TRAJECTORY PRINTOUT FOR TYPICAL 1981 LAUNCH/1984 ENCOUNTER

```
LAUNCH
   LAUNCH DATE    2444675.500 -   3/12/1981. 0. 0. 0
   C3                  56.473
   DLA                -24.402
   L-MAN. ANGLE       163.275
   DV1                  5.554
   PERIOD            1301.403
   PERIHELION            .993
   APHELION             3.672
   STATE          -.14663705E+09   .24405359E+08  -.59821025E+03
                  -.67235885E+01  -.36878121E+02  -.17283156E+00
   ELEMENTS        .34898042E+09   .57417124E+00   .26419107E+00
                  -.95080813E+01   .17763764E+03   .24207271E+01
   SPHERICAL       .14865428E+09  -.26911099E-03   .17055028E+03
                   .37486428E+02  -.26416371E+00   .25966737E+03

MANEUVER
   MANEUVER DATE 2445171.971 -   7/21/1982.11.18.14
   DELTA V              4.253
   DVSUM                9.807
   MAN.-T ANGLE       186.360
   PERIOD            1021.023
   PERIHELION            .341
   APHELION             3.628
   STATE           .47324004E+09  -.23259862E+09  -.59732839E+06
                   .59522524E+01   .43431006E+01   .13737860E+01
   ELEMENTS        .29685894E+09   .82815950E+00   .11923124E+02
                  -.25815393E+02  -.17405737E+03   .17369062E+03
   SPHERICAL       .52731275E+09  -.75769048E-01   .33382577E+03
                   .75032169E+01   .10549966E+02   .36070825E+02

TARGET
   ARRIVAL DATE   2445787.169 -   3/27/1984.16. 3.21
   VHP                   .343
   DVSUM               10.150
   PHASE               85.334
   SEV                -12.306
   I-S)VE              38.598
   RANGE                 .709
   STATE          -.47939887E+08   .17401751E+08  -.11004189E+07
                  -.22757439E+02  -.63536809E+02  -.14170703E+02
   ELEMENTS        .29685894E+09   .82815950E+00   .11923124E+02
                  -.25815393E+02  -.17405737E+03   .50706800E-01
   SPHERICAL       .51012397E+08  -.12360574E+01   .16004955E+03
                   .68964420E+02  -.11857509E+02   .25028564E+03
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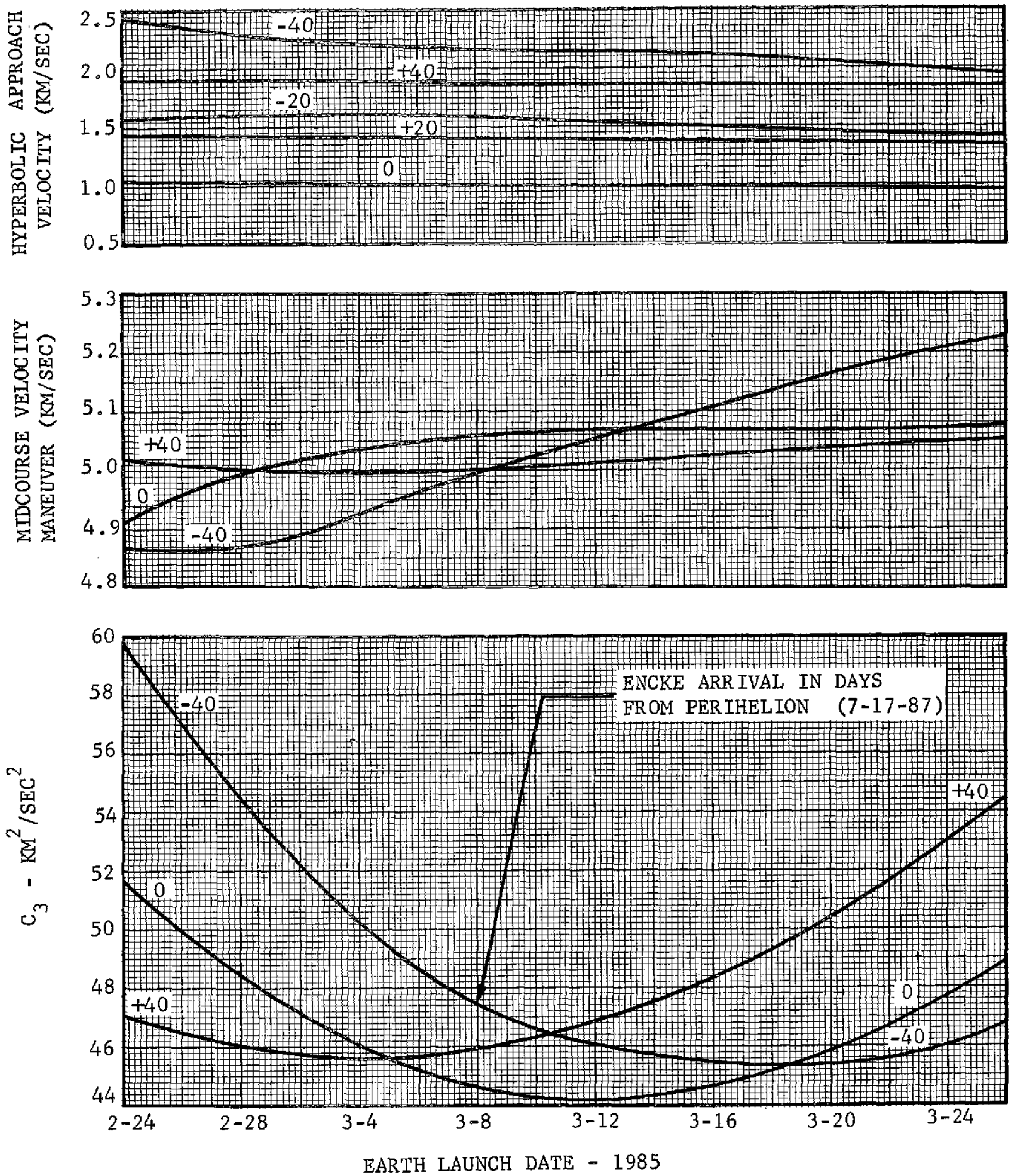

FIGURE III-5 PERFORMANCE PARAMETERS FOR 1985 LAUNCH/1987 ENCOUNTER

TABLE III-5 TRAJECTORY PRINTOUT FOR TYPICAL 1985 LAUNCH/1987 ENCOUNTER

```
LAUNCH
   LAUNCH DATE    2446135.500 -    3/11/1985. 0. 0. 0
   C3                  44.222
   DLA                -24.863
   L-MAN. ANGLE       164.164
   DV1                  5.089
   PERIOD            1026.547
   PERIHELION            .993
   APHELION             2.990
   STATE          -.14611539E+09    .27130844E+08    -.10074746E+04
                  -.67426982E+01   -.35987062E+02    -.18173590E+00
   ELEMENTS        .29792880E+09    .50118183E+00     .28439445E+00
                  -.10597190E+02    .17979904E+03     .27921717E+00
   SPHERICAL       .14861289E+09   -.38841880E-03     .16948106E+03
                   .36613735E+02   -.28439444E+00     .25938784E+03

MANEUVER
   MANEUVER DATE  2446535.768 -    4/15/1986. 6.25.55
   DELTA V              5.067
   DVSUM               10.156
   MAN.-T ANGLE       186.662
   PERIOD             769.920
   PERIHELION            .332
   APHELION             2.956
   STATE           .38652978E+09   -.19149805E+09    -.58148231E+06
                   .65810383E+01    .54494685E+01     .16386140E+01
   ELEMENTS        .24593690E+09    .79833698E+00     .11890509E+02
                  -.25988339E+02   -.17389455E+03     .17351970E+03
   SPHERICAL       .43136656E+09   -.77234758E-01     .33364485E+03
                   .87001051E+01    .10856167E+02     .39626662E+02

TARGET
   ARRIVAL DATE   2446993.954 -    7/17/1987.10.53.45
   VHP                   .994
   DVSUM               11.150
   PHASE               84.352
   SEV                -10.995
   (-S)VE             144.217
   RANGE                1.267
   STATE          -.46642185E+08    .16824565E+08    -.11190710E+07
                  -.22828943E+02   -.63945350E+02    -.14209177E+02
   ELEMENTS        .24593690E+09    .79833698E+00     .11890509E+02
                  -.25988339E+02   -.17389455E+03     .18157335E+00
   SPHERICAL       .49596489E+08   -.12929038E+01     .16016483E+03
                   .69369079E+02   -.11819808E+02     .25035308E+03
```

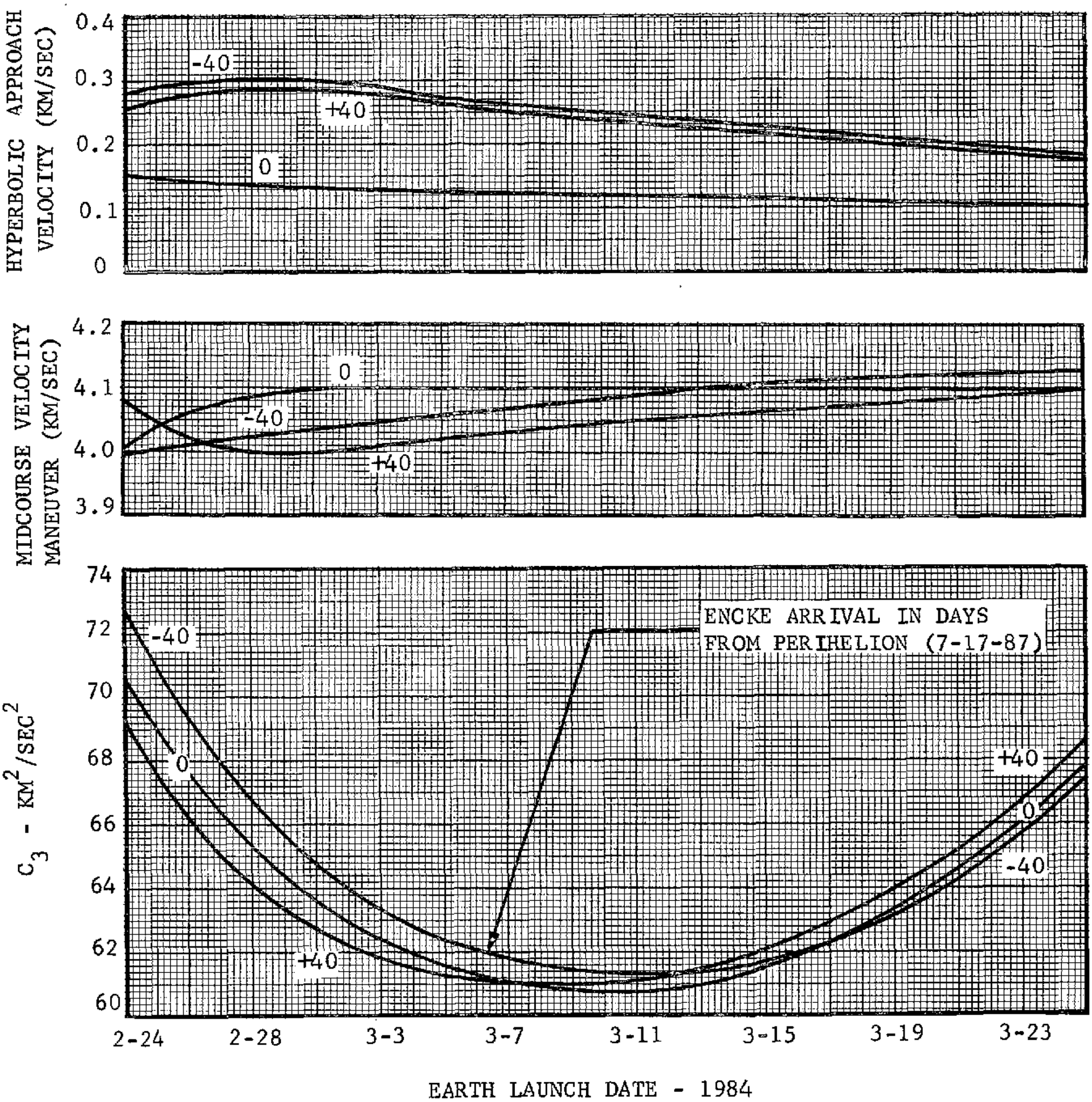

FIGURE III-6 PERFORMANCE PARAMETERS FOR 1984 LAUNCH/1987 ENCOUNTER

TABLE III-6 TRAJECTORY PRINTOUT FOR TYPICAL 1984 LAUNCH/1987 ENCOUNTER

LAUNCH			
LAUNCH DATE	2445770.500 -	3/11/1984. 0. 0. 0	
C3	60.724		
DLA	-23.517		
L-MAN. ANGLE	164.169		
DV1	5.712		
PERIOD	1417.435		
PERIHELION	.993		
APHELION	3.946		
STATE	-.14625056E+09	.26450571E+08	-.92550701E+03
	-.71737983E+01	-.37083507E+02	-.57198335E-01
ELEMENTS	.36942695E+09	.59777214E+00	.86768295E-01
	-.10487181E+02	.17837245E+03	.18631559E+01
SPHERICAL	.14862321E+09	-.35679250E-03	.16974842E+03
	.37771063E+02	-.86765481E-01	.25905138E+03
MANEUVER			
MANEUVER DATE	2446307.551 -	8/30/1985. 1.13.26	
DELTA V	4.101		
DVSUM	9.813		
MAN.-T ANGLE	186.384		
PERIOD	1126.170		
PERIHELION	.332		
APHELION	3.905		
STATE	.50783351E+09	-.24859243E+09	-.23019733E+06
	.58518201E+01	.38468287E+01	.12711547E+01
ELEMENTS	.31690497E+09	.84349729E+00	.11921184E+02
	-.25971979E+02	-.17374544E+03	.17363251E+03
SPHERICAL	.56541412E+09	-.23326860E-01	.33391753E+03
	.71174240E+01	.10288083E+02	.33319840E+02
TARGET			
ARRIVAL DATE	2446993.954 -	7/17/1987.10.53.45	
VHP	.124		
DVSUM	9.936		
PHASE	85.589		
SEV	-10.995		
(-S)VE	144.217		
RANGE	1.267		
STATE	-.46642185E+08	.16824565E+08	-.11190710E+07
	-.23026174E+02	-.64766721E+02	-.14421493E+02
ELEMENTS	.31690497E+09	.84349729E+00	.11921184E+02
	-.25971979E+02	-.17374544E+03	.16457640E-01
SPHERICAL	.49596489E+08	-.12929038E+01	.16016483E+03
	.70234695E+02	-.11848987E+02	.25042844E+03

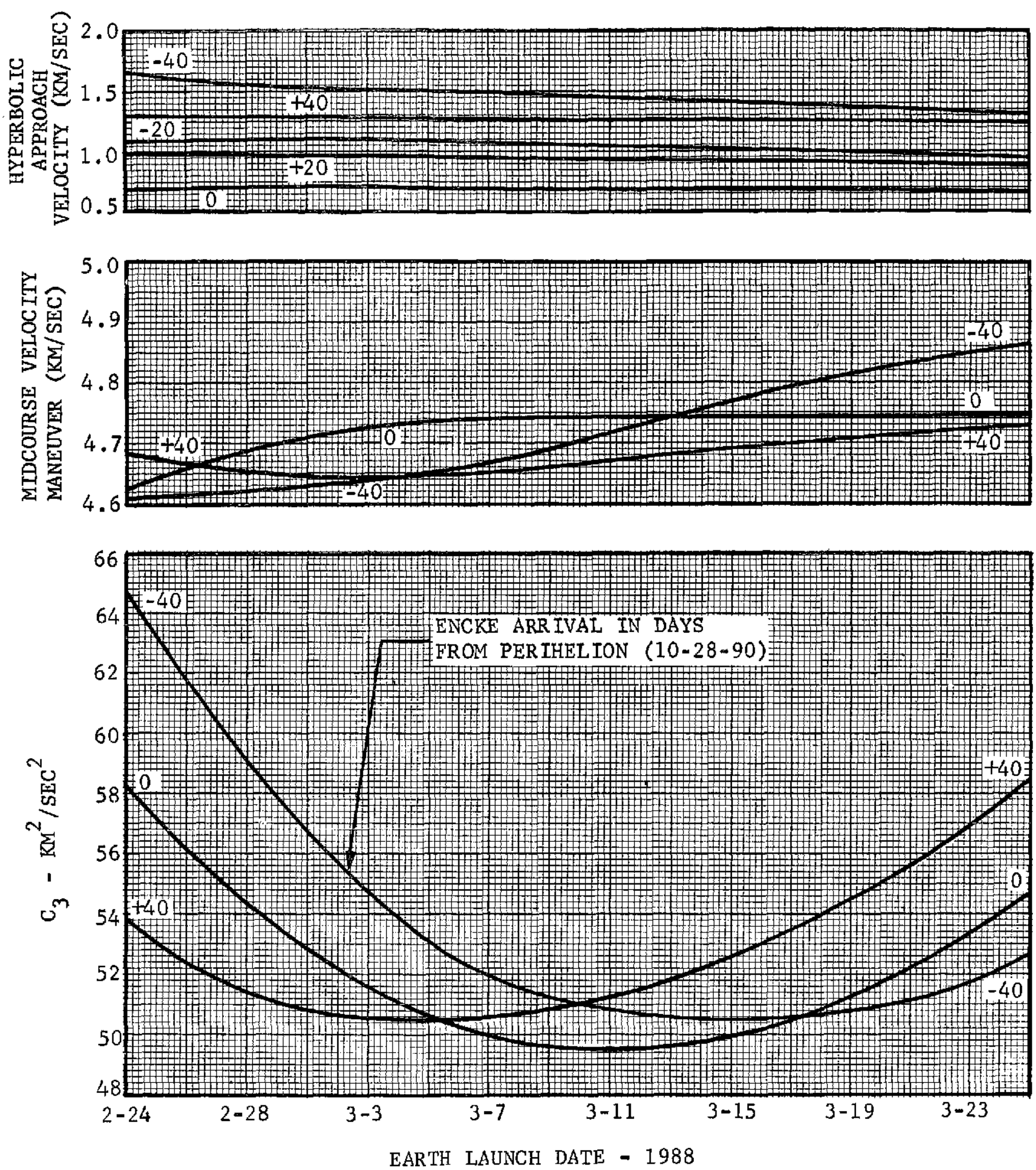

FIGURE III-7 PERFORMANCE PARAMETERS FOR 1988 LAUNCH/1990 ENCOUNTER

TABLE III-7 TRAJECTORY PRINTOUT FOR TYPICAL 1988 LAUNCH/1990 ENCOUNTER

```
LAUNCH
   LAUNCH DATE    2447231.500 -   3/11/1988. 0. 0. 0
   C3                  49.600
   DLA                -24.358
   L-MAN. ANGLE       164.072
   DV1                  5.295
   PERIOD            1137.975
   PERIHELION            .993
   APHELION             3.273
   STATE          -.14621627E+09    .26631408E+08    -.10475538E+04
                  -.68410309E+01   -.36376019E+02    -.13834683E+00
   ELEMENTS        .31911578E+09    .53429205E+00     .21415610E+00
                  -.10430579E+02    .17916513E+03     .94291574E+00
   SPHERICAL       .14862177E+09   -.40384671E-03     .16967747E+03
                   .37813965E+02   -.21415451E+00     .25934911E+03

MANEUVER
   MANEUVER DATE 2447671.130 -    5/24/1989.15. 7.11
   DELTA V              4.744
   DVSUM               10.039
   MAN.-T ANGLE       186.547
   PERIOD             370.267
   PERIHELION            .331
   APHELION             3.237
   STATE           .42263317E+09   -.20842344E+09    -.48016591E+06
                   .63180548E+01    .48749874E+01     .15086753E+01
   ELEMENTS        .26686690E+09    .81463503E+00     .11915422E+02
                  -.25973742E+02   -.17385109E+03     .17356832E+03
   SPHERICAL       .47123174E+09   -.58382070E-01     .33374958E+03
                   .81215405E+01    .10705574E+02     .37653643E+02

TARGET
   ARRIVAL DATE   2448192.967 -  10/28/1990.11.12.28
   VHP                   .686
   DVSUM               10.725
   PHASE               84.752
   SEV                 12.610
   (-S)VE            -139.011
   RANGE                1.219
   STATE          -.46518689E+08    .16787935E+08    -.11143704E+07
                  -.22936165E+02   -.64320627E+02    -.14321326E+02
   ELEMENTS        .26686690E+09    .81463503E+00     .11915422E+02
                  -.25973742E+02   -.17385109E+03     .11496688E+00
   SPHERICAL       .49467818E+08   -.12908215E+01     .15015621E+03
                   .69773284E+02   -.11844436E+02     .25037422E+03
```

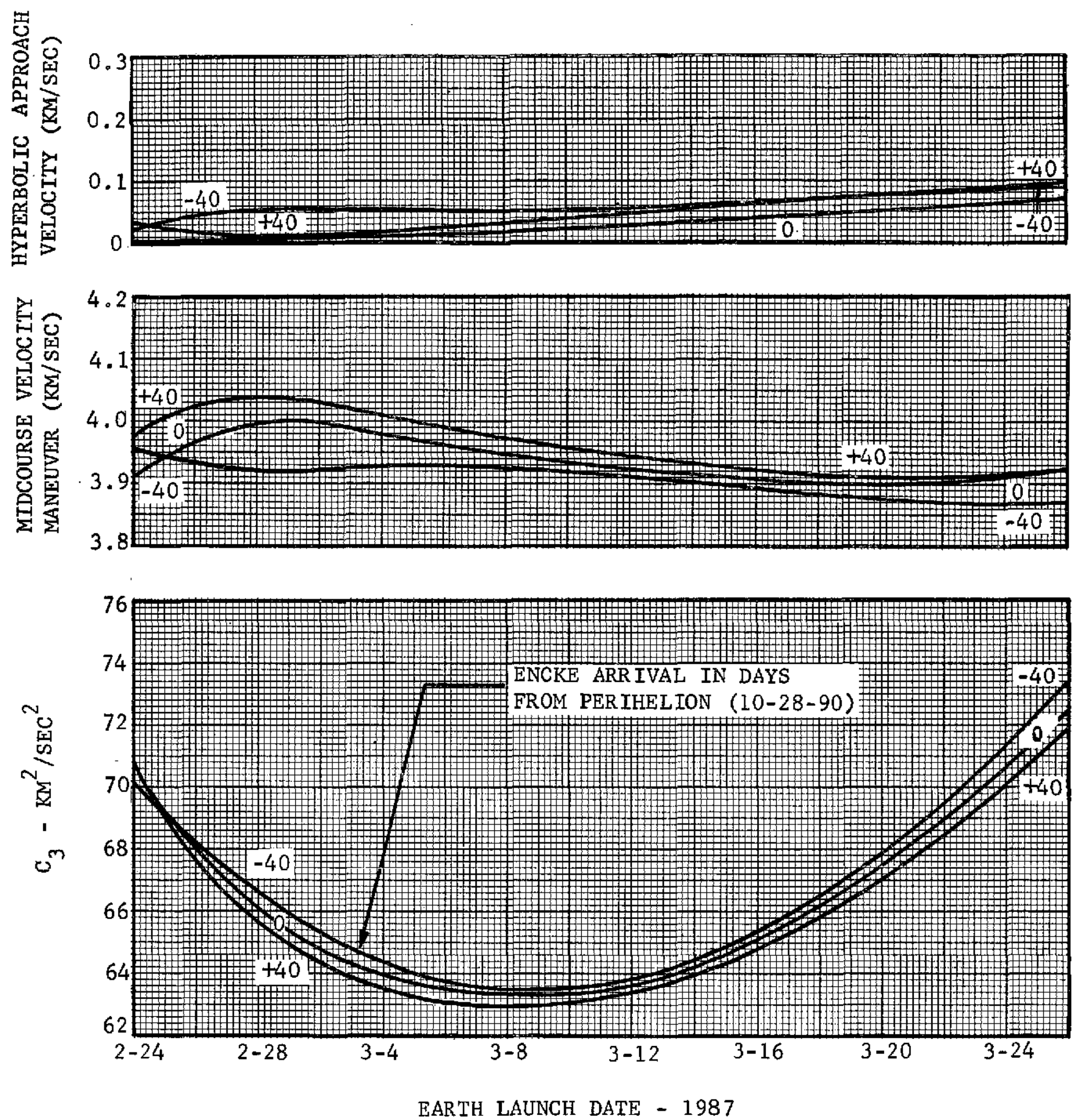

FIGURE III-8 PERFORMANCE PARAMETERS FOR 1987 LAUNCH/1990 ENCOUNTER

TABLE III-8 TRAJECTORY PRINTOUT FOR TYPICAL 1987 LAUNCH/1990 ENCOUNTER

```
LAUNCH
   LAUNCH DATE     2446866.500 -  3/12/1987. 0. 0. 0
   C3                   63.717
   DLA                 -21.748
   L-MAN. ANGLE        164.384
   DV1                   5.822
   PERIOD             1503.637
   PERIHELION             .993
   APHELION              4.144
   STATE          -.14634911E+09    .25950716E+08    -.96208715E+03
                  -.75462675E+01   -.37194649E+02     .15100740E+00
   ELEMENTS        .38425707E+09    .61350577E+00     .22804945E+00
                   .17003797E+03   -.38113331E+01     .37181539E+01
   SPHERICAL       .14863210E+09   -.37087233E-03     .16994479E+03
                   .37952746E+02    .22797059E+00     .25853117E+03

MANEUVER
   MANEUVER DATE  2447455.163 - 10/20/1988.15.54.16
   DELTA V               3.936
   DVSUM                 9.758
   MAN.-T ANGLE        185.955
   PERIOD             1217.409
   PERIHELION             .331
   APHELION              4.132
   STATE           .54037284E+09   -.25973332E+09     .64613208E+06
                   .54936543E+01    .36723741E+01     .12062007E+01
   ELEMENTS        .33379838E+09    .85180330E+00     .11934384E+02
                  -.25963677E+02   -.17374977E+03     .17404837E+03
   SPHERICAL       .59955368E+09    .61747012E-01     .33432847E+03
                   .67172531E+01    .10344572E+02     .33761763E+02

TARGET
   ARRIVAL DATE   2448192.967 - 10/28/1990.11.12.28
   VHP                    .029
   DVSUM                 9.786
   PHASE                85.952
   SEV                  12.610
   (-S)VE             -139.011
   RANGE                 1.219
   STATE          -.46518689E+08    .16787935E+08    -.11143704E+07
                  -.23110073E+02   -.64992499E+02    -.14488836E+02
   ELEMENTS        .33379838E+09    .85180330E+00     .11934384E+02
                  -.25963677E+02   -.17374977E+03     .38018190E-02
   SPHERICAL       .49467818E+08   -.12908215E+01     .16015621E+03
                   .70484231E+02   -.11862364E+02     .25042560E+03
```

C. COMPARISON OF MISSION OPTIONS

A summary of performance characteristics for all of the mission options investigated (including the 1980 ascending node case) is presented in Table III-9. No attempt has been made to achieve a fully optimized launch period with varying arrival times. A representative 10-day launch period, with a single selected arrival date and near-maximum performance capability, was selected for mission comparison purposes.

The 1980 ascending node case requires the highest C_3 (99.1 km^2/sec^2) of all the missions shown but does not require a midcourse velocity maneuver and has the shortest time of flight (less than 4 months). A final velocity maneuver of 2.88 km/sec is required to reduce the comet flyby velocity to 5.0 km/sec.

The descending node missions identified as potential slow flyby missions have the lowest C_3 requirements (30-50 km^2/sec^2), but require midcourse velocity maneuvers in the 5 to 6 km/sec range for simultaneously reducing perihelion and changing orbit inclination to a near-match with the comet plane. Approach velocities vary from 0.7 km/sec to 2.3 km/sec.

The potential rendezvous missions have higher C_3 requirements (50-65 km^2/sec^2) because of the higher aphelions achieved, but have lower midcourse velocity maneuvers (4 to 5 km/sec region) and approach velocities (from 40 m/sec to 350 m/sec).

TABLE III-9 SUMMARY OF PERFORMANCE CHARACTERISTICS

MISSION TYPE	CENTER OF 10-DAY LAUNCH PERIOD	ENCKE ENCOUNTER (1)	C_3 (KM^2/SEC^2)	ΔV_{MC} (KM/SEC)	V_{H_1} (2) (KM/SEC)	V_{H_2} (2) (KM/SEC)
(3)2.3 KM/SEC Flyby	3-02-79	12-16-80	32.8	5.79	2.29	2.29
(4)5.0 KM/SEC Flyby	8-23-80	12-12-80	99.1	0	7.88	5.00
(3)1.7 KM/SEC Flyby	3-02-82	04-06-84	40.5	5.27	1.66	1.66
1.0 KM/SEC Flyby	3-11-85	07-17-87	45.2	5.07	1.01	1.01
0.7 KM/SEC Flyby	3-11-88	10-28-90	50.3	4.74	0.70	0.70
Rendezvous	3-12-81	03-27-84	57.7	4.25	0.35	0
Rendezvous	3-11-84	07-17-87	62.0	4.10	0.13	0
Rendezvous	3-12-87	10-28-90	65.7	3.96	0.04	0

(1) Encounter at perihelion except as noted.

(2) Subscripts 1 and 2 designate approach velocity before and after, respectively, of the final maneuver (where applicable).

(3) Encounter at perihelion + 10 days.

(4) Encounter at perihelion + 5.5 days.

NOTE: Values of C_3, ΔV_{MC}, V_{H_1}, and V_{H_2} shown are the maximum occurring during the launch period.

Using the Titan IIIE/Centaur launch vehicle as a base, Table III-10 presents representative spacecraft characteristics for the missions summarized in Table III-9. Use of the TE364-4 upper stage would be required for the 1980 ascending node mission and desirable for the rendezvous missions launched in 1981, 1984 and 1987 (due to the C_3 requirements of these missions).

A two-stage, Earth-storable, bi-propellant system (equal propellant loads) was selected to provide the inflight velocity maneuvers, including the mid-course corrections. Because of the lower maneuver velocity required in the 1980 ascending node case, a single stage liquid propulsion system was used (A solid propellant alternate could also be considered).

Net spacecraft weight at encounter varies from a minimum of 245 kg for the 1979/80 flyby mission to a maximum of 340 kg for the 1987/90 rendezvous mission, with the Titan IIIE/Centaur as the basic launch vehicle. Availability of a Shuttle/Centaur (or equivalent) launch vehicle would permit increases of approximately 55 percent in the net spacecraft weight for any of the missions shown.

TABLE III-10 REPRESENTATIVE SPACECRAFT CHARACTERISTICS

LAUNCH VEHICLE: Titan IIIE/Centaur except as noted (Reference: NASA, Launch Vehicle Estimating Factors for Advance Mission Planning, NHB 7100.58, 1973 edition)

SPACECRAFT PROPULSION: Two-stage Earth storable bi-propellants except as noted:
Specific Impulse = 310 sec. Propellant Fraction = 0.9

MIDCOURSE CORRECTION ALLOWANCE: 150 m/sec

LAUNCH/ ENCOUNTER YEARS	MISSION TYPE	$C_3/\Delta V$ EFF (1) $(KM^2/SEC^2)/$ (KM/SEC)	INITIAL SPACECRAFT WEIGHT (KG)	(2) SPACECRAFT PROPELLANT WEIGHT (KG)	(3) NET SPACECRAFT WEIGHT AT ENCOUNTER (KG)
1979/1980	2.3 KM/SEC Flyby	32.8/5.94	2950	2435	245
1980/1980	5.0 KM/SEC Flyby	99.1/3.03	1000	635	(4)(5)295
1982/1984	1.7 KM/SEC Flyby	40.5/5.42	2600	2085	285
1985/1987	1.0 KM/SEC Flyby	45.2/5.22	2400	1900	290
1988/1990	0.7 KM/SEC Flyby	50.3/4.89	2170	1675	310
1981/1984	Rendezvous	57.7/4.75	1950	1490	(4)295
1984/1987	Rendezvous	62.0/4.38	1800	1330	(4)320
1987/1990	Rendezvous	65.7/4.15	1700	1230	(4)340

(1) ΔV Eff = Total effective velocity required by spacecraft propulsion system, including midcourse correction allowance.

(2) Total usable propellant required to produce ΔV Eff.

(3) Primary propulsion inert weight deducted.

(4) Launch Vehicle: Titan IIIE/Centaur/TE364-4

(5) Single stage spacecraft propulsion system used.

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IV. TRAJECTORY CHARACTERISTICS

IV. TRAJECTORY CHARACTERISTICS

The purpose of this section is to provide a detailed description of the transfer phase geometry and Encke encounter phase geometry for three representative cases. The three cases chosen for description are the 1980 slow flyby (ascending node case), the 1984 slow flyby, and the 1987 rendezvous mission. Also included in these results are the effects of launch date variation (across a 10-day launch period) upon the important mission parameters.

The information provided should be useful in assessing the environment of the spacecraft throughout the mission, assist in identifying Earth communication, navigation, and tracking problems, and furnish insight into science observation opportunities during the encounter phase.

A. 1980 LAUNCH/1980 ENCOUNTER MISSION

Figure IV-1 presents the heliocentric transfer phase geometry for a typical 1980 (ascending node launch) trajectory. Total time of flight from launch to encounter is 111 days. Data are actually shown out to 220 days and reflect the 2.88 km/sec velocity maneuver made just before comet encounter. Spacecraft range from the Sun varies from 1 AU at launch to a minimum of 0.34 AU a few days before comet encounter. The Earth communication range at encounter is about 1.2 AU and approaches 1.6 AU at 20 days after encounter.

The spacecraft equatorial declination is within a few degrees of zero for the first 70 days of the flight and reaches a maximum negative value of -28 deg after Encke encounter. (Positive declination is measured North of the Earth's equator).

The Sun-Earth-spacecraft angle is defined as the angle between the vector from the Earth to the Sun and the vector from Earth to the spacecraft. Looking downward from the northern ecliptic pole, the angle is measured positive clockwise going from the Earth-Sun vector to the Earth-spacecraft vector and lies between +180 deg and -180 deg. In Figure IV-1 and the corresponding figures for the other missions shown, the Sun-Earth-spacecraft angle has a folded scale for convenience of presentation. The dashed line indicates positive values of the angle, as defined.

For this typical 1980 mission trajectory, the Sun-Earth-spacecraft angle reaches a minimum value of 13 deg at 33 days before encounter and is increased to 16 deg away from the Sun at encounter. No serious navigation or tracking

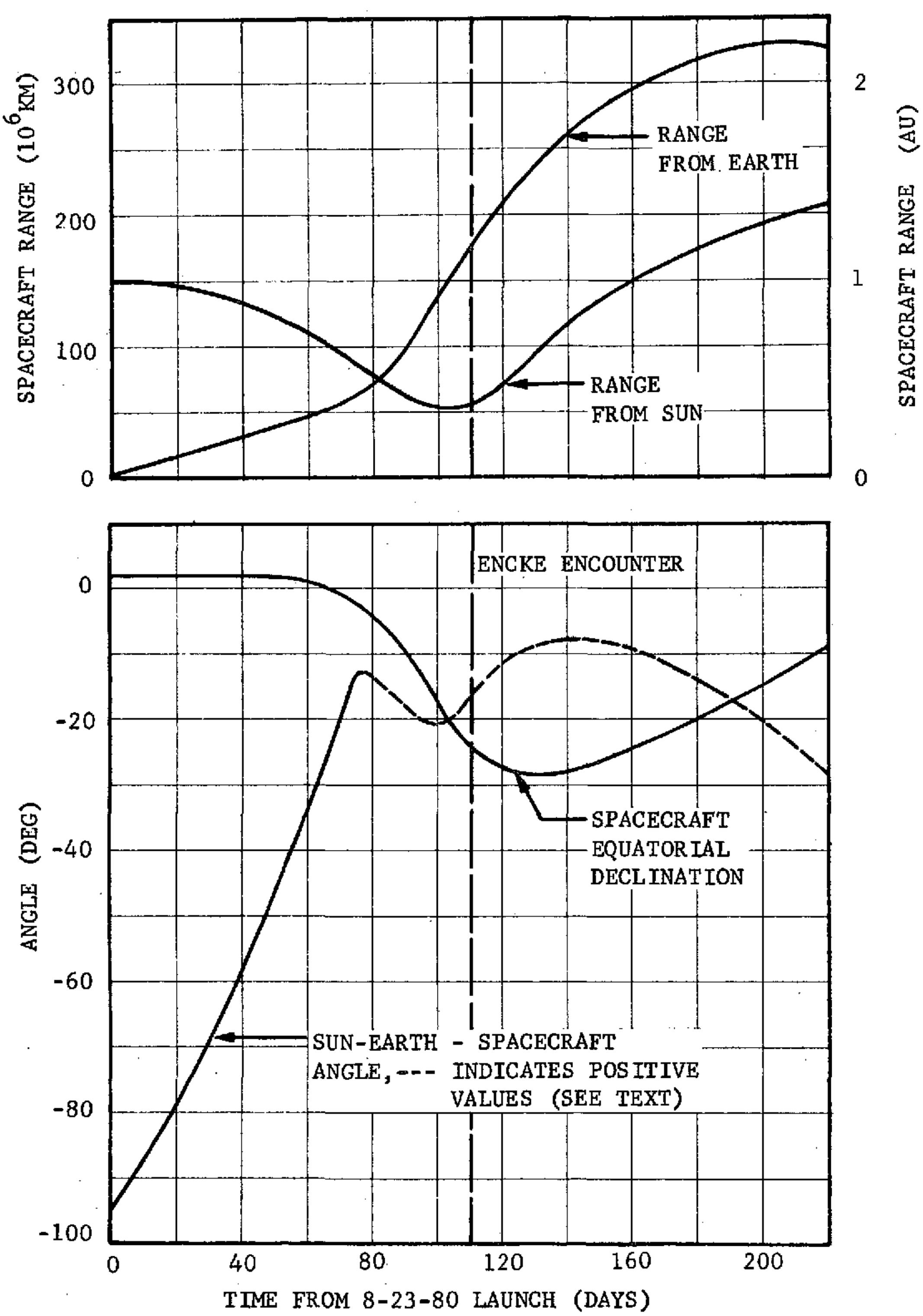

FIGURE IV-1 TRANSFER PHASE PARAMETERS FOR TYPICAL 1980 LAUNCH/1980 ENCOUNTER

and communications problems are anticipated.

Figure IV-2 presents typical comet encounter geometry for a launch in the center of the 10-day launch period. In this figure, as in the corresponding figures for the other missions described, the motion shown is the relative motion between the spacecraft and the comet projected on the comet plane. The positive downrange direction is always toward the Sun (measured from the comet) with the positive crossrange axis to the left (looking from the comet to the Sun).

The post-encounter trajectory reflects the impulsive velocity maneuver of 2.88 km/sec at encounter and can be observed in the shortening intervals between 2-day time ticks. Note that the approach phase angle is 118 deg, or nearly a cross-tail motion. (The approach phase angle is the angle between the hyperbolic-approach-velocity vector and the comet-position vector at encounter and lies between 0 deg and +180 deg).

To show the effect of varying launch date throughout the 10-day period, Table IV-1 summarizes important mission parameters for comparison. The reference date corresponds to the middle of the launch period and also corresponds to the data shown in Figures IV-1 and IV-2.

For a fixed arrival date of 5.5 days after Encke perihelion passage, the maximum approach velocity of 7.88 km/sec occurs at the beginning of the launch period, while the maximum C_3 requirement (99.1 km^2/sec^2) is reached at the end of the period. Launch declinations are low in all cases and due East launches could always be used (no launch azimuth penalty). All of the trajectories are Type II with heliocentric transfer angles varying from 220 to 230 deg. Approach phase angles remain within 5 deg of the reference value.

NOTES

LAUNCH = 8-23-80

ENCKE ARRIVAL = 12-12-80

ALL MOTION IS RELATIVE TO ENCKE AND PROJECTED UPON THE COMET PLANE

TIME TICKS IN DAYS FROM ENCOUNTER

118 DEG PHASE ANGLE AT ENCOUNTER

APPROACH VELOCITY = 7.67 KM/SEC BEFORE ENCOUNTER

DEPARTURE VELOCITY = 4.79 KM/SEC AFTER MANEUVER

ΔV AT ENCOUNTER = 2.88 KM/SEC

--- INDICATES RELATIVE MOTION AFTER MANEUVER

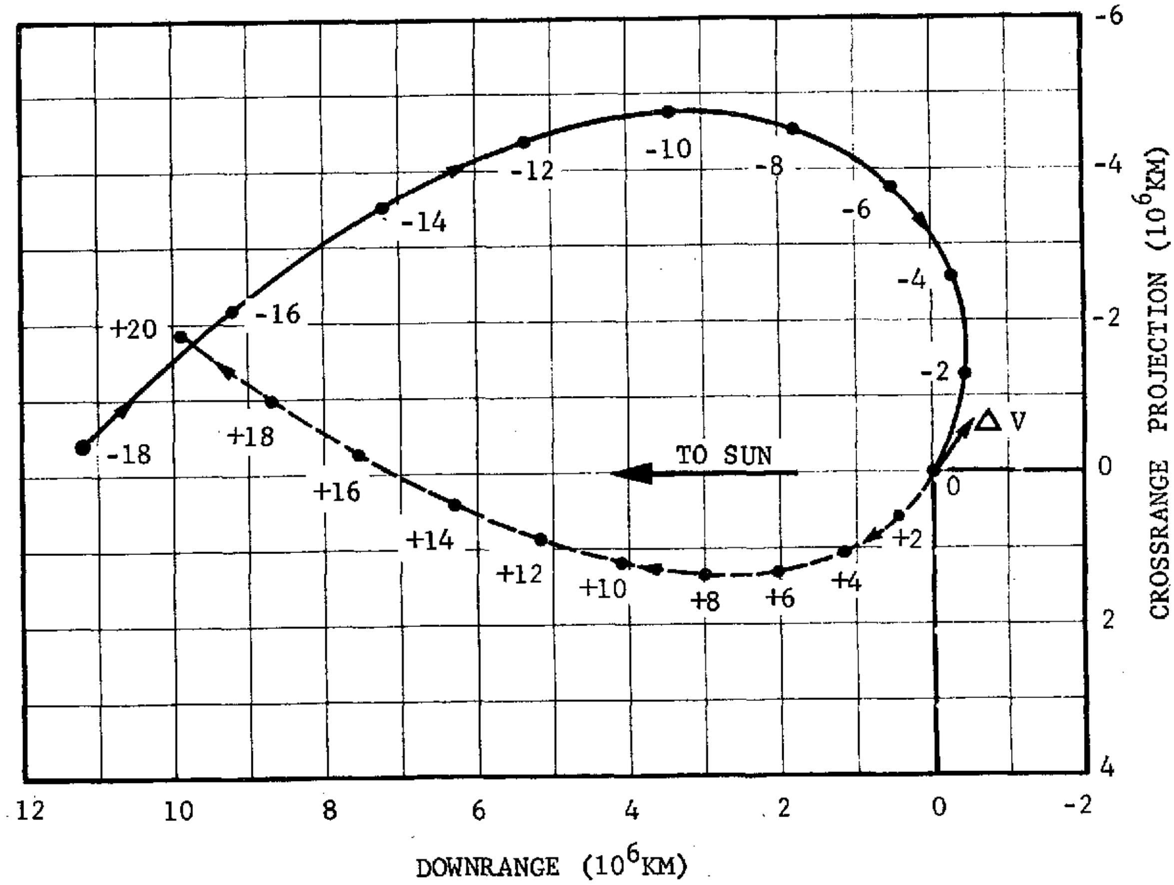

FIGURE IV-2 ENCOUNTER-PHASE GEOMETRY FOR TYPICAL 1980 LAUNCH/1980 ENCOUNTER

TABLE IV-1 LAUNCH DATE EFFECTS FOR 1980 LAUNCH/1980 ENCOUNTER

	EARTH LAUNCH (DAYS FROM REFERENCE)*		
	-5	0	+5
C_3 (KM^2/SEC^2)	97.2	96.2	99.1
DLA (DEG)	-0.1	2.2	4.7
Orbit Inclination (DEG)	11.51	10.92	11.94
Transfer Angle (DEG)	230.3	225.6	220.9
Perihelion (AU)	0.33	0.34	0.34
Trip Time (DAYS)	116.0	111.0	106.0
Approach Phase Angle (DEG)	112.4	117.9	122.1
Initial Approach Velocity (KM/SEC)	7.88	7.67	7.65
Final Approach Velocity (KM/SEC) **	5.00	4.79	4.77

* Reference: Earth Launch 8-23-80; Encke Arrival 12-12-80

** 2.88 KM/SEC Final Velocity Maneuver before Encounter.

B. 1982 LAUNCH/1984 ENCOUNTER MISSION

Typical heliocentric and Earth-centered spacecraft time histories are presented in Figure IV-3 for the 1000-day period from launch until over 200 days after comet encounter. Spacecraft maximum range from the Sun is 2.6 AU and occurs near the main maneuver about 1 year after launch. Range from the Earth is also at a maximum at this time (3.55 AU), and the spacecraft is in solar occultation within a week before the maneuver (Sun-Earth-spacecraft angle is zero). This region was further investigated during the navigation analysis and is discussed in Section V.

Earth range is greatly reduced during the encounter period (reduced to 0.75 AU) and the Sun-Earth-spacecraft angle, although diminishing is still over 10 deg in magnitude at encounter. Cycles in both the range from Earth and the Sun-Earth-spacecraft angle can be observed in Figure IV-3 as the Earth goes through 2 full revolutions about the Sun during the mission.

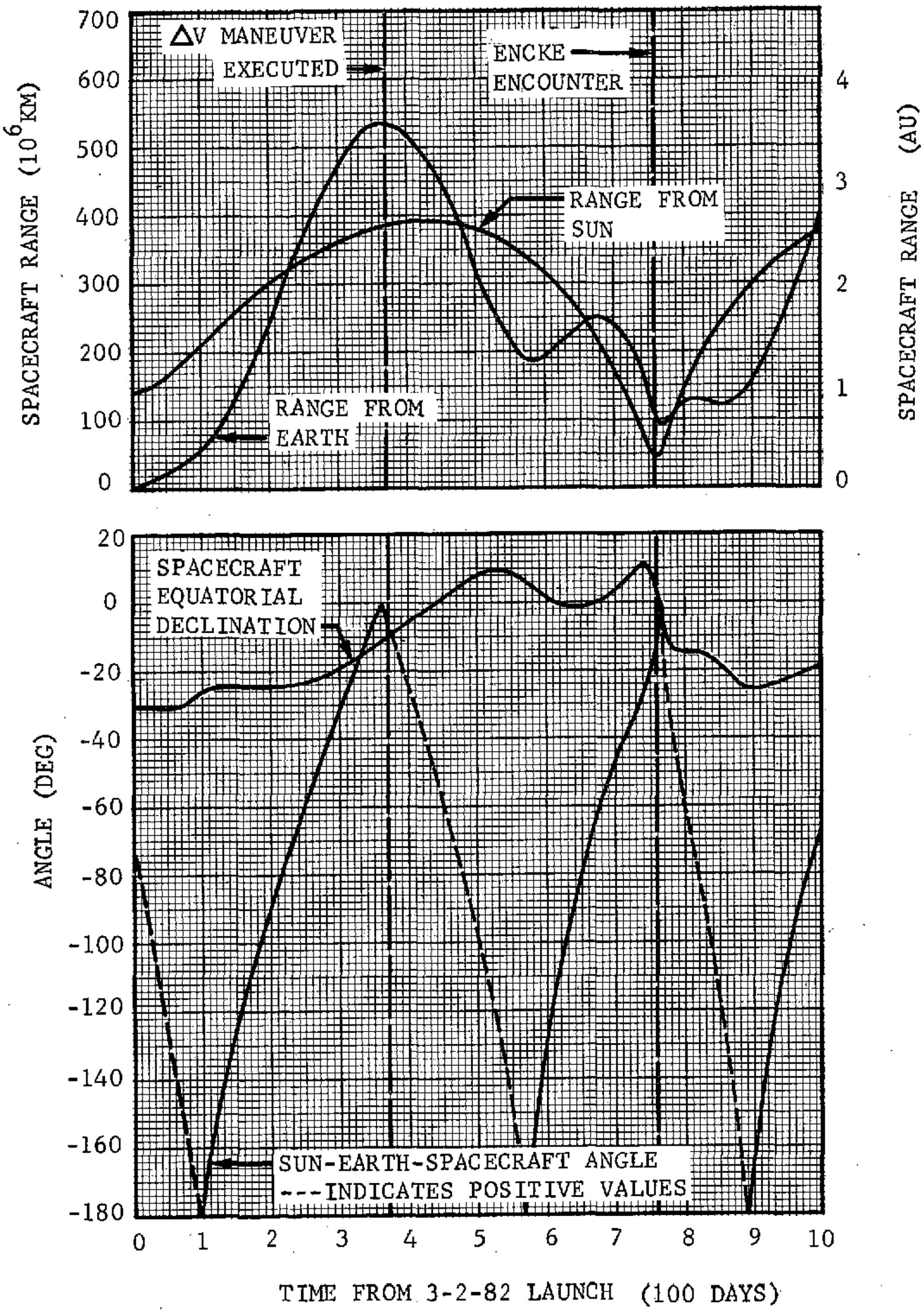

FIGURE IV-3 TRANSFER PHASE PARAMETERS FOR TYPICAL 1982 LAUNCH/1984 ENCOUNTER

Figure IV-4 shows typical encounter geometry starting 18 days before arrival at the comet (comet is at perihelion) and ending 20 days after encounter. Because of the substantially reduced flyby velocity (1.46 km/sec), the spacecraft stays within close proximity for many days. The approach phase angle is 81 deg.

Table IV-2 summarizes the effects of varying time of launch through the 10-day launch period. The reference launch date is 3-2-82 with Encke arrival on 3-27-84. Spacecraft (S/C) orbit characteristics are shown for both orbit segments of the mission. Note that the transfer times before and after the planned midcourse velocity maneuver (5.37 km/sec maximum) are about equal and are approximately 1 year each.

The effect of the ΔV_{MC} optimization is also shown by the distribution of orbit inclination angles between the pre-maneuver and post-maneuver orbits, with the major plane change in the final orbit segment. Approach phase angle and approach velocity do not vary significantly through the launch period.

NOTES

LAUNCH = 3-2-82

MANEUVER DATE = 3-6-83

ENCKE ARRIVAL = 3-27-84

ALL MOTION IS RELATIVE TO ENCKE AND PROJECTED UPON THE COMET PLANE

TIME TICKS IN DAYS FROM ENCOUNTER

81 DEG PHASE ANGLE AT ENCOUNTER

APPROACH VELOCITY = 1.46 KM/SEC

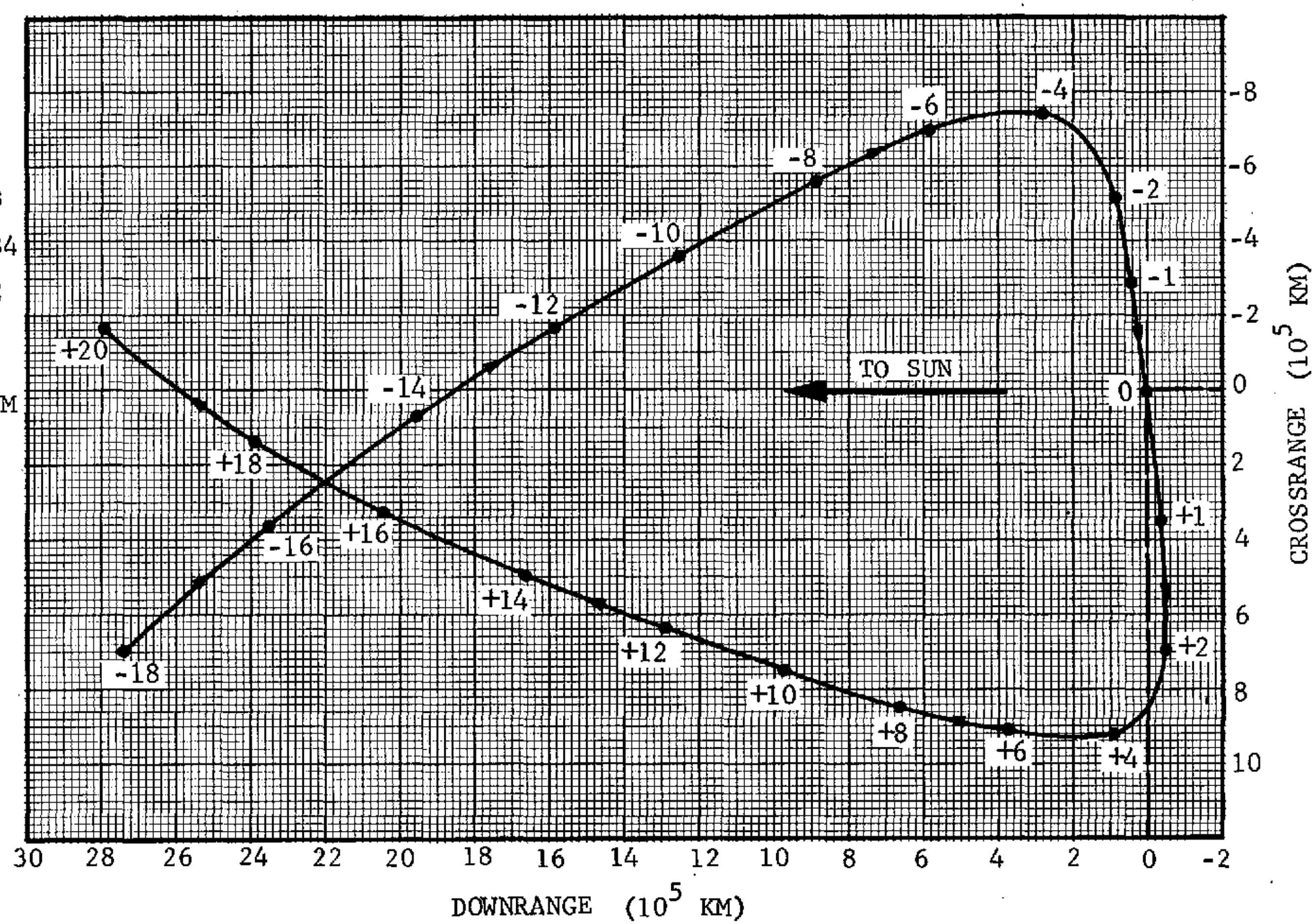

FIGURE IV-4 ENCOUNTER PHASE GEOMETRY FOR TYPICAL 1982 LAUNCH/1984 ENCOUNTER

TABLE IV-2

LAUNCH DATE EFFECTS FOR 1982 LAUNCH/1984 ENCOUNTER

	EARTH LAUNCH (DAYS FROM REFERENCE)*		
	-5	0	+5
C_3 (KM^2/SEC^2)	44.0	40.3	38.4
DLA (DEG)	-37.1	-31.4	-28.1
Pre-Maneuver S/C Orbit:			
Inclination (DEG)	2.48	1.39	0.80
Transfer Angle (DEG)	178.6	173.1	168.1
Aphelion (AU)	2.63	2.65	2.66
Transfer Time (DAYS)	380.3	369.2	362.5
ΔV_{MC} (KM/SEC)	5.20	5.32	5.37
Post-Maneuver S/C Orbit:			
Inclination (DEG)	11.87	11.88	11.87
Transfer Angle (DEG)	186.3	186.8	186.8
Perihelion (AU)	0.34	0.34	0.34
Transfer Time (DAYS)	381.4	387.5	389.2
Total Trip Time (DAYS)	761.7	756.7	751.7
Approach Phase Angle (DEG)	79.9	81.0	82.4
Approach Velocity (KM/SEC)	1.47	1.46	1.45

* Reference: Earth Launch 3-2-82; Encke Arrival 3-27-84

C. 1984 LAUNCH/1987 ENCOUNTER MISSION

Figure IV-5 presents typical heliocentric and Earth-centered spacecraft time histories for a representative Comet Encke rendezvous mission launched on 3-11-84 and arriving at the comet 3.3 years later during the 1987 perihelion passage (7-17-87). The cyclic behavior of the Earth-related parameters is similar to that shown in Figure IV-3.

In this case, however, the Earth is on the same side of the sun as the spacecraft during the primary velocity maneuver of 4.1 km/sec and solar occulation is not a consideration. Range from the Earth to the spacecraft is about 2.8 AU at this time. A maximum Earth range of about 4.9 AU occurs

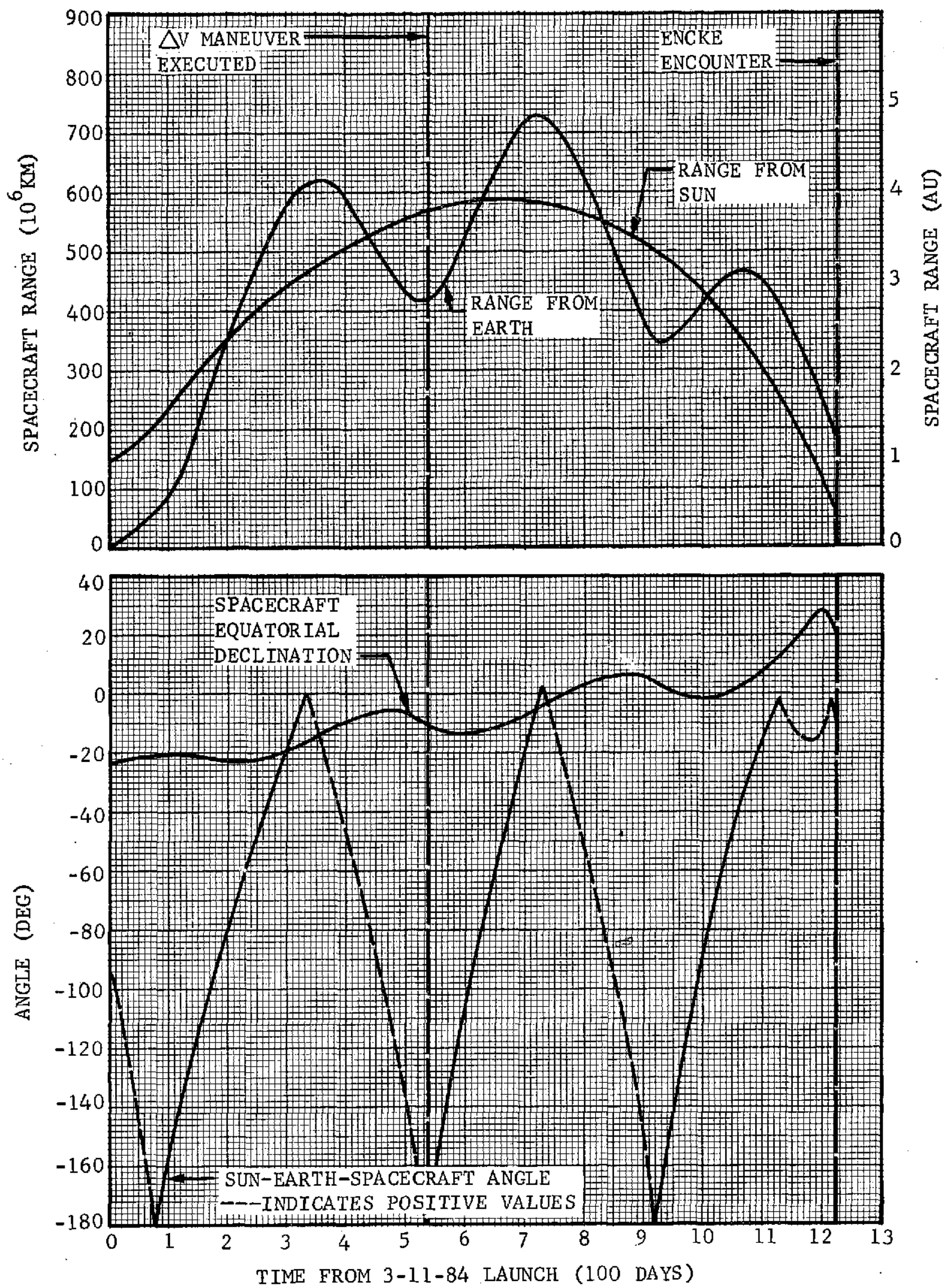

FIGURE IV-5 TRANSFER PHASE PARAMETERS FOR TYPICAL 1984 LAUNCH/1987 ENCOUNTER

during the second orbit segment of this mission but is reduced to about 1.2 AU at Encke encounter. A near solar occultation occurs within two weeks of encounter and is further considered in the navigation analysis (Section V).

Figure IV-6 shows typical encounter geometry starting at 22 days before encounter and ending at encounter (rendezvous). Notice the greatly reduced scale (all motion shown within 300,000 km down range distance) due to the very low approach velocity of 124 m/sec. The spacecraft actually stays within close proximity of the comet for many months due to the near-match of the comet and spacecraft orbits at the time of the primary velocity maneuver. Approach phase angle is 85.6 deg.

The launch date effects are summarized in Table IV-3. A launch C_3 of over 60 is required to provide an initial spacecraft orbit aphelion of about 4 AU (the optimum orbit inclination for the first orbit segment is near zero). The total trip time of 3.3 years is divided between the first and second orbit segments in a .45 to .55 ratio, respectively. The approach phase angle varies from 82.0 deg to 87.5 deg over the launch period, and the approach velocity shows no significant variation.

NOTES

EARTH LAUNCH = 3-11-84

MANEUVER DATE = 8-30-85

ENCKE ARRIVAL = 7-17-87

ALL MOTION IS RELATIVE TO ENCKE AND PROJECTED UPON THE COMET PLANE

TIME TICKS IN DAYS FROM ENCOUNTER

85.6 DEG PHASE ANGLE AT ENCOUNTER

APPROACH VELOCITY BEFORE ENCOUNTER AND RENDEZVOUS = 124 M/SEC

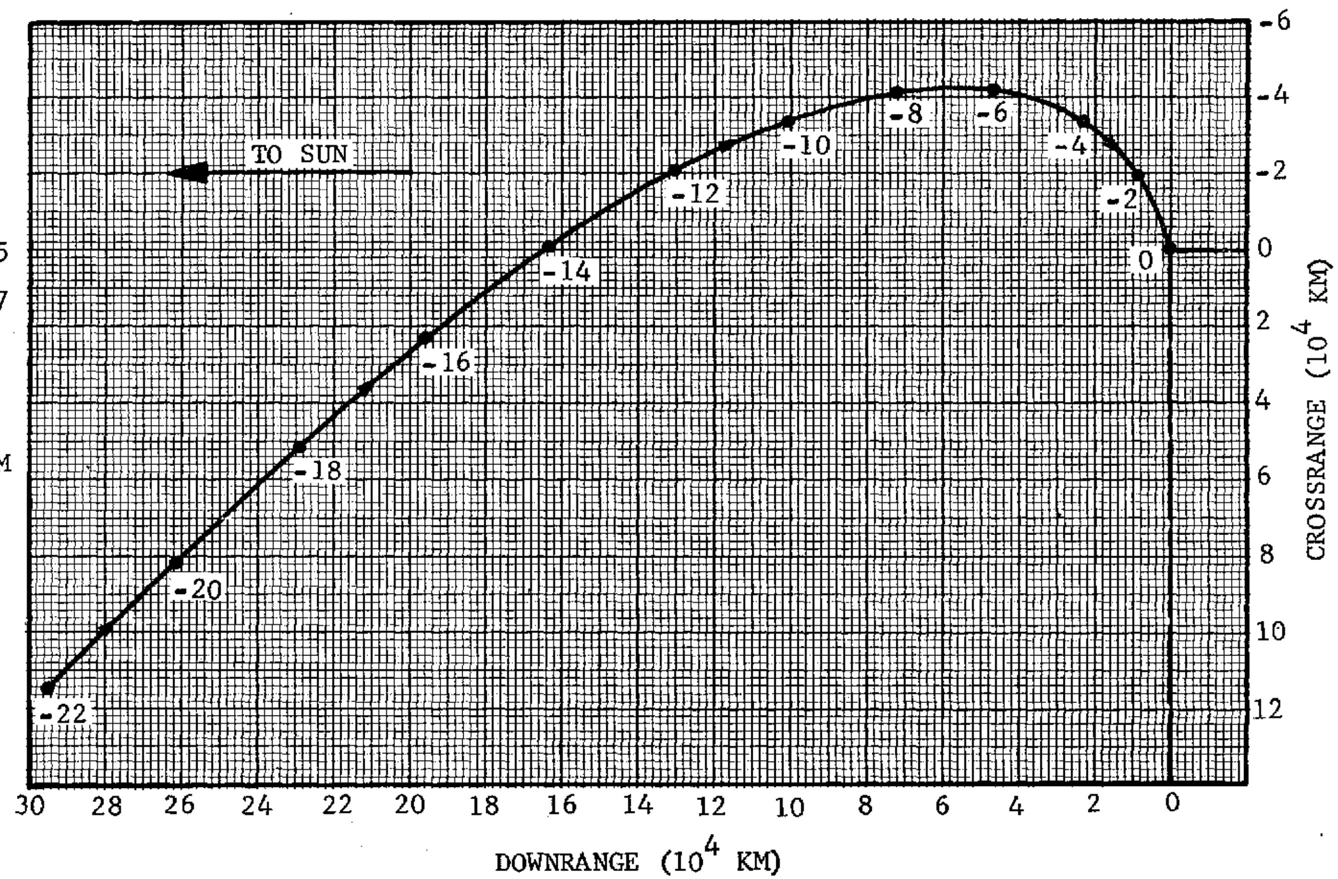

FIGURE IV-6 ENCOUNTER PHASE GEOMETRY FOR TYPICAL 1984 LAUNCH/1987 ENCOUNTER

TABLE IV-3

LAUNCH DATE EFFECTS FOR 1984 LAUNCH/1987 ENCOUNTER

	EARTH LAUNCH (DAYS FROM REFERENCE)*		
	-5	0	+5
C_3 (KM^2/SEC^2)	61.4	60.7	62.0
DLA (DEG)	-24.2	-23.5	-23.6
Pre-Maneuver S/C Orbit:			
Inclination (DEG)	0.2	0.1	0.1
Transfer Angle (DEG)	169.1	164.2	159.1
Aphelion (AU)	3.93	3.95	3.98
Transfer Time (DAYS)	542.8	537.1	526.1
ΔV_{MC} (KM/SEC)	4.10	4.10	4.09
Post-Maneuver S/C Orbit:			
Inclination (DEG)	11.92	11.92	11.92
Transfer Angle (DEG)	186.4	186.4	186.5
Perihelion (AU)	0.33	0.33	0.33
Transfer Time (DAYS)	685.7	686.4	692.4
Total Trip Time (DAYS)	1228.5	1223.5	1218.5
Approach Phase Angle (DEG)	82.0	85.6	87.5
Initial Approach Velocity (M/SEC)	130 **	124	114
Final Approach Velocity (M/SEC)	0	0	0

* Reference: Earth Launch 3-11-84; Encke Arrival 7-17-87

** Maximum Final Maneuver Velocity for Rendezvous = 130 M/SEC

V. NAVIGATION ANALYSIS

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V. NAVIGATION ANALYSIS

Navigation analyses have been performed for two representative missions of the descending-node class. The missions chosen for study were the 1982 launch/1984 encounter flyby mission and the 1984 launch/1987 encounter rendezvous mission. (A navigation analysis for two missions of the 1980 ascending-node class was reported in the earlier phase of this contract).

The 1982/84 flyby mission is characterized by a low launch energy requirement (C_3 = 40 km^2/sec^2), a short time of flight (2.1 yrs), a large midcourse maneuver velocity (5.32 km/sec) and an encounter velocity of 1.46 km/sec. The 1984/87 rendezvous mission has a large C_3 required (61 km^2/sec^2), a substantially longer time of flight (3.3 yrs), a much lower midcourse velocity maneuver (4.10 km/sec), and a very low encounter velocity (124 m/sec). The heliocentric-phase geometry and encounter phase geometry for these missions are illustrated in Figures IV-3 through IV-6.

The primary results of interest in this analysis are:

(1) Assessment of the navigation feasibility of these missions.

(2) Determination of the total ΔV budget for the trim maneuvers.

(3) Evaluation of dispersions at comet encounter.

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A. ANALYSIS DESCRIPTION

The conic trajectories upon which the 82/84 mission and 84/87 mission navigation analyses were based were presented in Table III-3 and Table III-6, respectively. In both cases, the target for injection was the position state of the midcourse velocity maneuver given in the tables. The target for the post-maneuver orbit segment was the position state of the comet at perihelion for each encounter (same as target position states given in Tables III-3 and III-6).

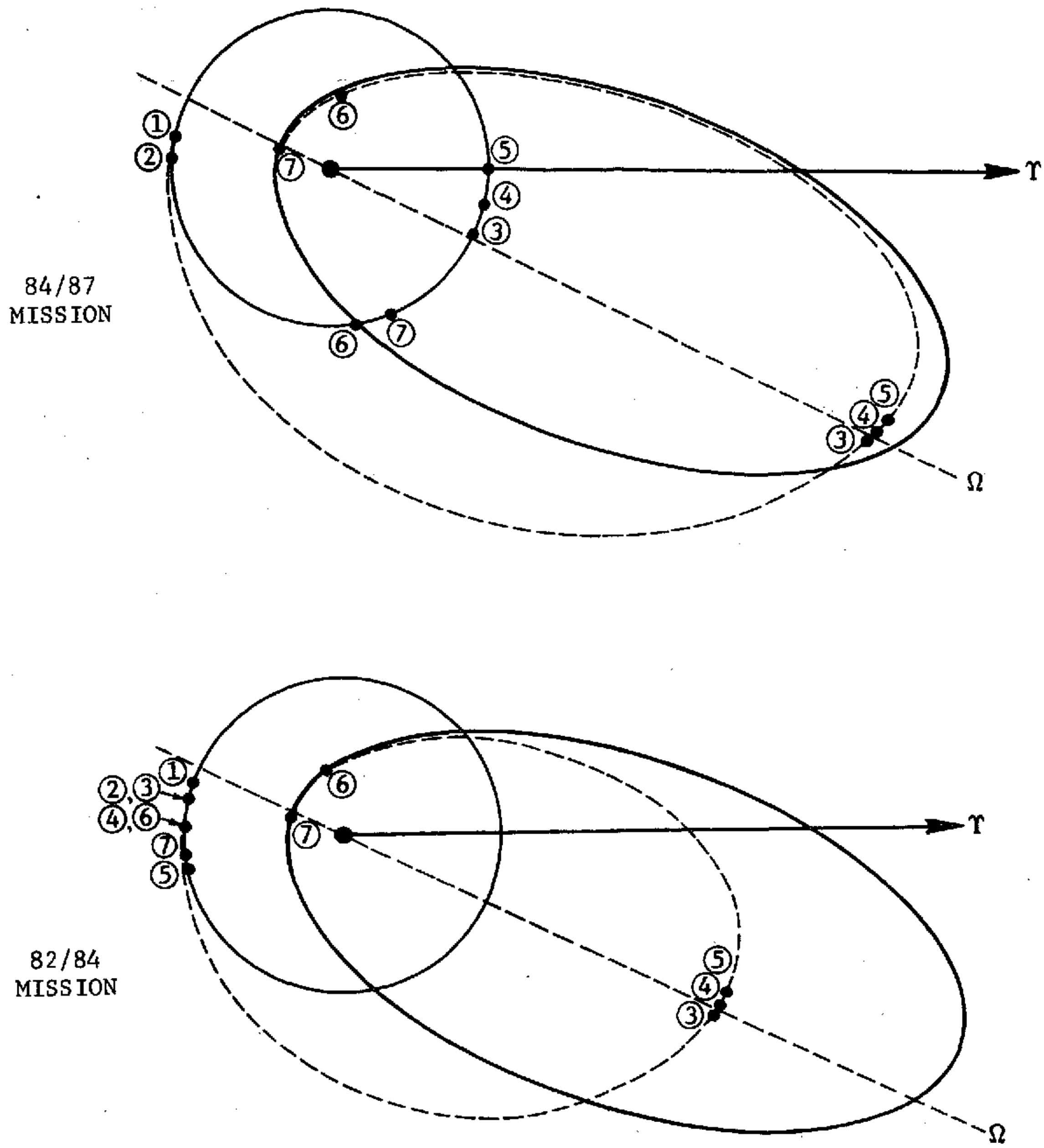
N-body targeting was used throughout the analysis. The planets considered were Mercury, Venus, Earth, Mars, Jupiter, and Saturn.

The maneuver strategies selected for the two missions were similar and are indicated on the heliocentric scale drawings of the complete trajectories from injection to encounter shown in Figure V-1.

The first trim maneuver (δV_1) was scheduled for 5 days after injection. (An additional maneuver considered at 25 days after injection was found to be unnecessary). The second trim maneuver (δV_2) was executed at 10 days after the planned midcourse velocity maneuver (ΔV_{MC}), with a third trim maneuver (δV_3) at 25 days after the ΔV_{MC} event. (A trim maneuver to reduce position dispersions prior to ΔV_{MC} was unnecessary since the contribution of these dispersions was small compared to the ΔV_{MC} execution errors). The maneuver strategy for both missions was identical up to this point.

At 10 days before Encke encounter, the final trim maneuver (δV_4) was executed for the 82/84 flyby mission. For the 84/87 rendezvous mission, it was necessary to schedule the final trim maneuver at 15 days before comet encounter because of the low Sun-Earth-spacecraft angles encountered shortly after this time (shown also in Figure IV-5).

The tracking schedules were also similar for both missions. A general policy of limiting tracking periods (consisting of one or more tracking arcs near the same event) to 30 days or less was adhered to throughout the analysis. Tracking arcs were generally initiated at 0.5 days after an event or trim maneuver and terminated at 0.5 days before a trim maneuver. A detailed tracking and maneuver schedule is shown in Table V-1.



EVENT CODE FOR EARTH AND SPACECRAFT POSITIONS

① Injection
② Injection + 5 Days, δV_1
③ Planned Midcourse Maneuver (ΔV_{MC})
④ ΔV_{MC} + 10 Days, δV_2
⑤ ΔV_{MC} + 25 Days, δV_3
⑥ Encounter - X Days, δV_4 (X=10 for 82/84; X=15 for 84/87)
⑦ Encke Encounter

FIGURE V-1 MANEUVER STRATEGIES

TABLE V-1 TRACKING AND MANEUVER SCHEDULE

CONDITION	TIME RELATIVE TO EVENT (DAYS) (1)	MANEUVER TIMES (FROM INJECTION) 82/84 MISSION (DAYS)	84/87 MISSION (DAYS)
Injection	L + 0	0	0
Earth Tracking	L + 0.5 to L + 4.5		
First Trim Maneuver	L + 5	5.0	5.0
Earth Tracking	L + 5.5 to L + 30		
Midcourse Velocity Maneuver	M + 0	368.4	536.2
Earth Tracking	M + 0.5 to M + 9.5		
Second Trim Maneuver	M + 10	378.4	546.2
Earth Tracking	M + 10.5 to M + 24.5		
Third Trim Maneuver	M + 25	393.4	561.2
Earth Tracking	M + 25.5 to M + 30		
Resume Earth Tracking	E - 25		
Stop Earth Tracking (82/84)	E - 10.5		
Stop Earth Tracking (84/87)	E - 15.5		
Fourth Trim Maneuver (2)	E -10/E - 15	745.8	1207.6
Resume Earth Tracking (82/84)	E - 9.5		
Resume Earth Tracking (3) (84/87)	E - 14.5		
Encke Encounter	E + 0	755.8	1222.6

(1) L = Injection Point; M = Midcourse velocity maneuver event; E = Encounter with comet.

(2) At E - 10 for 82/84 mission; At E - 15 for 84/87 mission.

(3) Tracking is continuous to encounter for 82/84 mission; Tracking is interrupted between E - 13 and E - 5 on 84/87 mission (near solar occultation).

To simplify the analysis, injection errors associated with the Centaur stage were used in both cases. (The total ΔV budget for the trim maneuvers would actually be slightly higher for the 84/87 rendezvous mission because of the larger injection errors associated with the TE364-4 upper stage, but would have a negligible effect on the performance estimate).

The execution error model used for both the planned midcourse maneuver and the trim maneuvers was as follows (one-sigma values):

$\sigma_{Proportionality}$ = 1/3%

$\sigma_{Resolution}$ = 3×10^{-3} m/sec

$\sigma_{Pointing}$ = 1/3 deg.

Further assumptions were also made relative to dynamic noise (spacecraft outgassing and other unmodeled accelerations), measurement noise, and station location errors, as follows:

(1) Dynamic noise was assumed to perturb the flight with a spherical acceleration dispersion (Radius = 3×10^{-6} mm/sec^2).

(2) A measurement noise variance was assumed, equivalent to the JPL value of 1 mm/sec for a 1 minute measurement time.

(3) The equivalent station location errors assumed were as follows:

σ_{RS} = 4.05 meters (distance from Earth spin axis)

σ_{λ} = 3.70 meters (longitude)

σ_{Z} = 10 meters (Z-height)

$\sigma_{\lambda\lambda}$ = 0.9 (longitude correlation between stations)

B. MIDCOURSE CORRECTION REQUIREMENTS

A summary of the individual and total velocity budgets for the four midcourse correction maneuvers (for both missions) is shown in Table V-2. For each individual maneuver, the mean value, one-sigma deviation level, and 99 percentile level are given. The total velocity budgets shown (99 percentile level) were determined by statistically combining the individual budgets.

In both missions, the second trim maneuver (δV_2) is the main contributor to the total velocity budget and is due to the execution errors associated with the primary midcourse velocity maneuver. The δV_2 value of 74.9 m/sec for the 84/87 mission compared to 103.2 m/sec for the 82/84 mission is closely related to the correspondence in the magnitudes of the planned midcourse maneuver velocities (4.10 km/sec for the 84/87 mission compared to 5.32 km/sec for the 82/84 mission).

The analysis shows that the last midcourse correction maneuver (δV_4) could be substantially reduced (in both cases) by performing the maneuvers at an earlier time (e.g. E-20 to E-30 days) without significantly affecting the dispersions. Reduction in the total δV budget would of course be limited due to the large contribution of δV_2.

The first and third midcourse correction maneuvers are small and have a minor effect upon the total δV budget. The resulting total $\delta V_{.99}$ budget is 120.0 m/sec for the 82/84 mission and 94.7 m/sec for the 84/87 mission.

As previously mentioned, the presence of the TE364-4 upper stage on the 84/87 mission and also the 81/84 and 87/90 rendezvous missions would increase the δV_1 maneuver, but this effect would be offset by reduced δV_2 levels associated with lower planned midcourse maneuvers (as seen in the 84/87 mission). For performance calculations, a conservative 150 m/sec allowance was made for midcourse correction maneuvers for all missions.

TABLE V-2

SUMMARY OF THE VELOCITY BUDGETS FOR THE MIDCOURSE CORRECTION MANEUVERS

A. 1982/84 FLYBY MISSION

TRIM MANEUVER	TIME RELATIVE TO EVENT (DAYS)	INDIVIDUAL δV VALUES (M/SEC) MEAN	ONE-SIGMA	99 PERCENTILE
δV_1	L + 5	2.2	1.1	5.6
δV_2	M + 10	46.0	20.8	103.2
δV_3	M + 25	1.1	0.7	3.4
δV_4	E - 10	10.0	7.3	31.5

Total Budget ($\delta V_{.99}$ Total) = 120.0 m/sec

Equivalent f sigma level = 2.743 sigma

B. 1984/87 RENDEZVOUS MISSION

TRIM MANEUVER	TIME RELATIVE TO EVENT (DAYS)	MEAN	ONE-SIGMA	99 PERCENTILE
δV_1	L + 5	2.6	1.4	6.7
δV_2	M + 10	33.5	15.1	74.9
δV_3	M + 25	0.9	0.6	2.8
δV_4	E - 15	11.3	7.5	33.9

Total Budget ($\delta V_{.99}$ Total) = 94.7 m/sec

Equivalent f sigma level = 2.740 sigma

NOTE: The $\delta V_{.99}$ total value is computed from the equation

$$\delta V_{.99} \text{ total} = m + f\sigma$$

Where m is the sum of the means, σ is the RSS value of the individual sigmas, and f is the sigma level corresponding to the 99 percentile value of the largest δV.

C. ENCOUNTER DISPERSIONS

The projected target dispersions (at both the planned maneuver point and Encke encounter) produced by the assumed trim-maneuver and tracking schedules are presented in Table V-3 for both missions investigated. The trajectory control and knowledge are projected forward in time to the target from the conditions immediately before and after each trim maneuver.

The large dispersions before the δV_1 and δV_3 maneuvers reflect the long flight times and trajectory sensitivities of the two missions. The higher dispersions for the 84/87 mission (compared to the 82/84 mission) are mainly a function of the longer flight times. Projected dispersions are greatly reduced after the trim maneuvers in both cases.

As previously mentioned, the final trim maneuvers could be made at an earlier time if desired without significantly affecting the final dispersions shown. The encounter dispersions for the 82/84 mission and the 84/87 mission are 72.2 km and 234.5 km, respectively (one-sigma values of the semimajor axes of the dispersion ellipsoids). The larger encounter dispersions associated with the 84/87 mission are caused jointly by the longer flight time and the long Earth tracking distances near encounter (see Figure V-1).

The total actual dispersions at Encke will be a function of both the above dispersions and the dispersions due to Encke ephemeris uncertainties, as indicated in the earlier phase of this contract. Section VI will report on some recent work in the area of estimating Comet Encke ephemeris uncertainties. The dominating ephemeris uncertainties are then compared to the dispersions associated with the navigation trim maneuvers.

Both missions investigated (and the other missions of this class) appear feasible from a navigation standpoint. Standard tracking techniques are satisfactory for the schedules selected.

TABLE V-3 PROJECTED TARGET DISPERSIONS

A. 1982/84 FLYBY MISSION

TRIM MANEUVER	TIME RELATIVE TO EVENT (DAYS)	TARGET * POINT	SEMI-MAJOR AXIS OF DISPERSION ELLIPSOID (KM)	
			BEFORE MANEUVER	AFTER MANEUVER
δV_1	L + 5	M	122,453	1316
δV_2	M + 10	E	-	46506
δV_3	M + 25	E	46510	9886
δV_4	E - 10	E	10273	72.2

B. 1984/87 RENDEZVOUS MISSION

TRIM MANEUVER	TIME RELATIVE TO EVENT (DAYS)	TARGET * POINT	BEFORE MANEUVER	AFTER MANEUVER
δV_1	L + 5	M	141,578	2268
δV_2	M + 10	E	-	76087
δV_3	M + 25	E	76099	15139
δV_4	E - 15	E	18221	234.5

* M corresponds to the position state of planned midcourse velocity maneuver (ΔV_{MC}). E corresponds to the position state of Encke at perihelion.

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VI. ENCKE EPHEMERIS ANALYSIS

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VI. ENCKE EPHEMERIS ANALYSIS

Prediction of ephemeris errors for comets presents a unique problem due to the limitations of optical data, non-gravitational effects, etc. MMC has conducted in-house investigations of Comet Encke (for which non-gravitational effects are negligible) to determine the applicability of spacecraft tracking-data processing methods to the type of data obtainable from comet observations. Since such information is deemed significant to the objectives of this study contract, available results are presented in this section.

A. ANALYSIS DESCRIPTION

The primary conditions and assumptions which were employed for the investigation are listed below.

1. 1984 apparition of Encke.
2. Tracking commenced 300 days prior to 1984 perihelion passage and continued through perihelion.
3. Observation frequency: one per week.
4. Observation data type: right ascension and declination.
5. Observation data quality: 3 arc-seconds measurement noise,
   3 arc-seconds measurement bias.
6. Observation data processing: Kalman-Schmidt estimation algorithm.
7. A priori knowledge covariances based on 1980 apparition.

		CORRELATIONS					
		x	y	z	vx	vy	vz
x	9921 km						
y	10112 km	-.52					
z	3894 km	-.07	-.09				
vx	.4 m/s	.89	-.86	.02			
vy	.2 m/s	-.43	-.54	.16	.04		
vz	.1 m/s	.27	.13	.94	.09	-.39	

B. ERROR PREDICTIONS

Preliminary results of the foregoing tracking program are presented on Figure VI-1 for the 40 day interval preceding perihelion. As shown, the predominant error component is oriented along the comet velocity vector. The relatively small cross-track errors also display assymmetry with the best confidence in the direction normal to the orbit plane. The rapid improvement in prediction during the last 15 days of observation is due to the particular tracking geometry associated with the 1984 apparition. Other Encke passage years would exhibit different characteristics; however, the indicated magnitudes and distributions are believed to be representative.

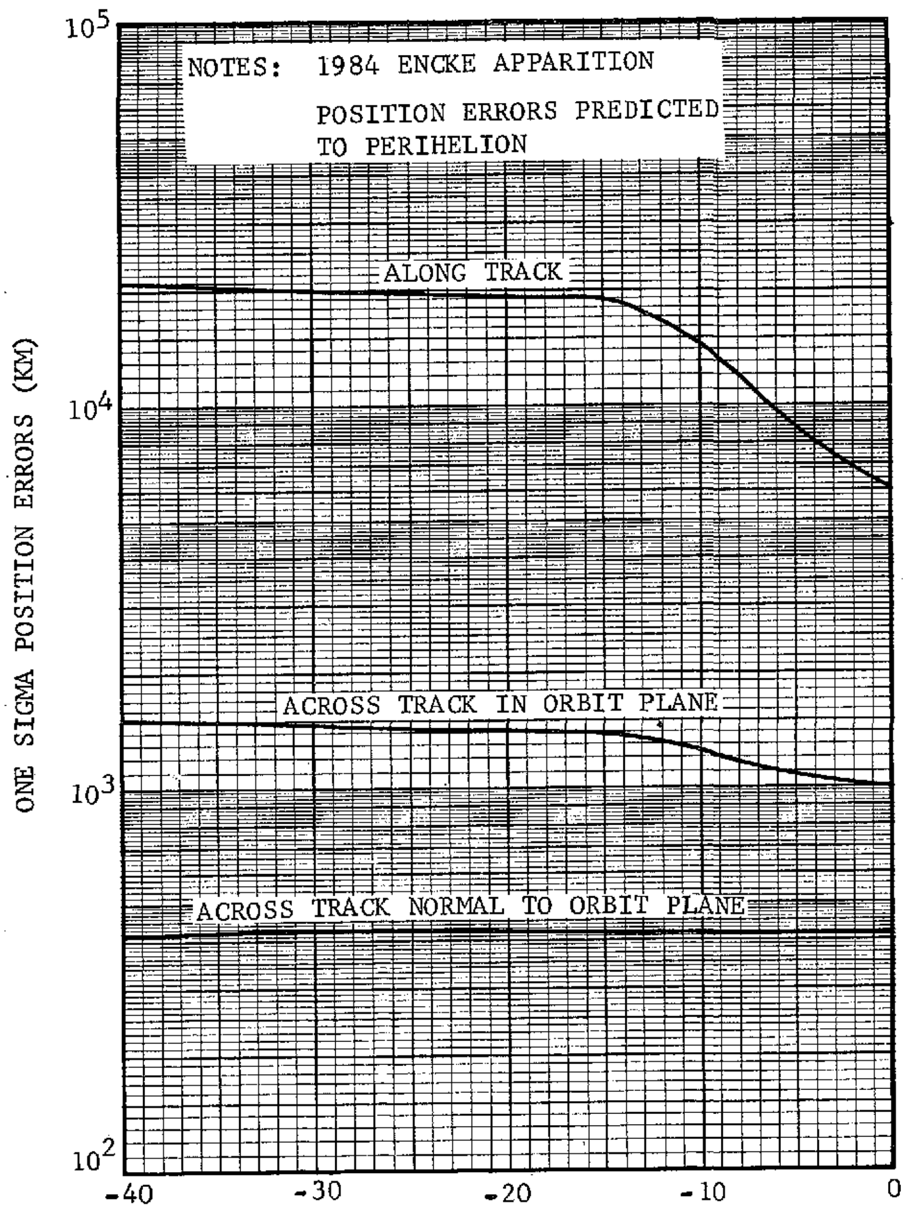

FIGURE VI-1 POSITION ERROR HISTORY FOR PERIHELION ENCOUNTER

C. ENCOUNTER IMPLICATIONS

For Encke slow flyby and rendezvous missions, best performance generally corresponds to encounter near the comet perihelion. Under these conditions, the relative velocity at approach is nearly tangential to the comet velocity. For these reasons, the primary ephemeris error component (along the comet track) does not directly affect nucleus miss distance but rather produces uncertainty in the time of closest approach.

Figure VI-2 displays the ephemeris error components directly contributing to the approach encounter dispersions. For comparison, the equivalent dispersions attributable to spacecraft navigation (see Section V) are superimposed. As shown, the comet ephemeris errors clearly predominate. However, the expected magnitudes would not impose severe requirements on spacecraft systems such as instrument pointing, attitude control and propulsion.

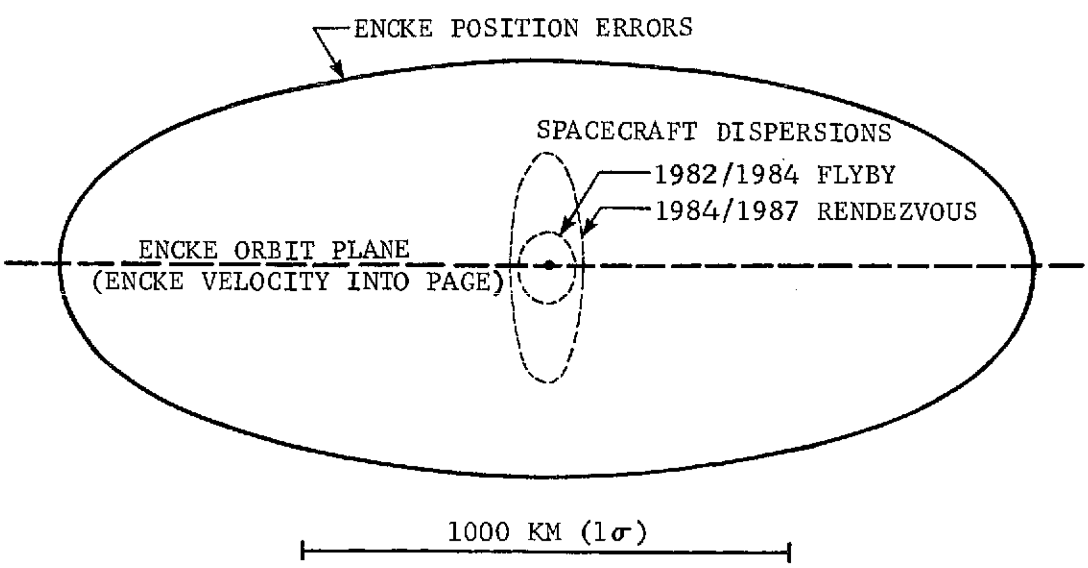

FIGURE VI-2 EPHEMERIS ERROR EFFECTS ON PERIHELION ENCOUNTER DISPERSIONS

VII. APHELION-CLASS RENDEZVOUS OPTIONS

VII. APHELION-CLASS RENDEZVOUS OPTIONS

Due to the geometry of the Encke orbit, best performance for rendezvous missions generally corresponds to encounter near the comet perihelion. For encounter substantially before or after perihelion, relative approach velocity usually increases rapidly and severely degrades performance potential. There are, however, exceptions to this behaviour which permit consideration of rendezvous as far out as aphelion. This section presents preliminary data for mission opportunities which could significantly reduce rendezvous trip time as well as support the ultimate comet science objective of sample return.

A. CANDIDATE MISSIONS

The rationale for selecting Encke mission opportunities for detailed performance analysis was described in Section II and was predicated on the assumption of near-perihelion encounter. Representative data were presented to illustrate the dependence of mission velocity requirements on time-of-flight.

Figure VII-1 repeats a portion of Figure II-3 and displays the unique condition for which the final rendezvous maneuver vanishes. This is the result of orbit phasing, which corresponds to positioning the large midcourse maneuver on the Encke orbit, thus eliminating the requirement for a subsequent maneuver at comet perihelion. Accordingly, the indicated time-of-flight of 3.7 years (referenced to perihelion for plotting consistency) is too high by half the Encke orbit period. For this special case, rendezvous was actually complete 2.05 years after launch and occurred at Encke aphelion (for the simplified analysis depicted). As shown by Figure VII-1, the timing of the 1987 launch to perihelion of the 1990 Encke apparition is nearly perfect for aphelion-class rendezvous. Consequently, performance is insensitive to Encke encounter date for nearly two years prior to the arrival span presented in Section III. The implications of this option include the prospects of initiating meaningful science data return much earlier in the mission and monitoring comet activity during the descending half-orbit (as opposed to the outbound phase dictated by perihelion encounter).

The other two mission cases depicted on Figure VII-1 (1984 and 1990 launch) represent different types of mis-alignment from the ideal for aphelion-class rendezvous. However, due to the small magnitudes of the final rendezvous maneuver (at perihelion), the spacecraft and comet orbits must be in near proximity for some time prior to encounter. In view of these indications, the 1984 and 1990 mission opportunities were also investigated for early rendezvous potential.

CONDITIONS: LAUNCH IN ECLIPTIC FROM ENCKE DESCENDING NODE
CO-PLANAR ENCOUNTER AT ENCKE PERIHELION
ENCKE LINE OF APSIDES ASSUMED COINCIDENT WITH ENCKE LINE OF NODES

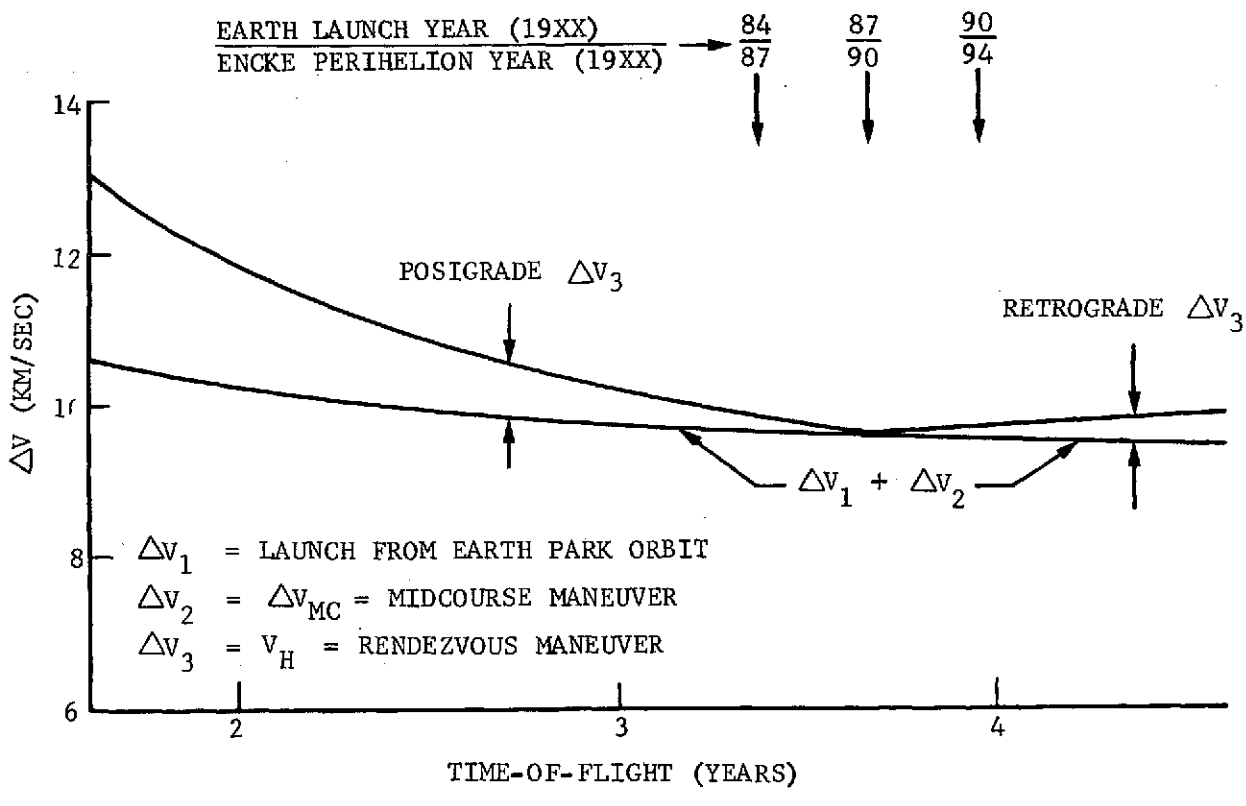

FIGURE VII-1 APHELION-CLASS RENDEZVOUS MISSION CANDIDATES

B. 1984 EARLY RENDEZVOUS MISSION

The region of best performance presented in Section III for the 1984 launch/1987 encounter mission is characterized by total flight time of about 40 months. Arrival 40 days before perihelion results in modest degradation of performance due primarily to an increase in the final rendezvous maneuver of about 150 mps. This trend was further investigated to evaluate the penalties associated with significant reductions in trip time.

Figure VII-2 presents data for early arrival extending through the region of Encke aphelion. For convenience of presentation, the two post-launch velocity maneuvers have been summed to characterize spacecraft propulsion requirements. The heliocentric radius at rendezvous is indicated to provide a perspective for the associated implications to communications and tracking (including Encke observation).

To facilitate interpretation of Figure VII-2, representative spacecraft weights have been calculated on a basis consistent with the values presented in Tables I-1 and III-10 for perihelion rendezvous. Table VII-1 summarizes performance requirements and capabilities for a wide range of flight time. As shown, the rate of performance degradation is non-linear with flight time reduction. Consequently, the prospects of early rendezvous for the 1984 launch opportunity appear encouraging.

TABLE VII-1 PERFORMANCE SENSITIVITY TO TIME-OF-FLIGHT FOR 1984 LAUNCH

TIME-OF-FLIGHT (MONTHS)	C_3 (KM^2/SEC^2)	$\Delta V_{MC} + V_H$ (KM/SEC)	NET S/C WEIGHT AT RENDEZVOUS (KG)
40.2	62.0	4.23	320
36	62.8	4.45	290
30	64.0	4.66	260
24	65.9	4.77	240
18	69.5	4.84	220

NOTES: NEXT ENCKE PERIHELION DATE, 7-17-87

ALL LAUNCH-TO-ΔV_{MC} TRAJECTORIES ARE TYPE I

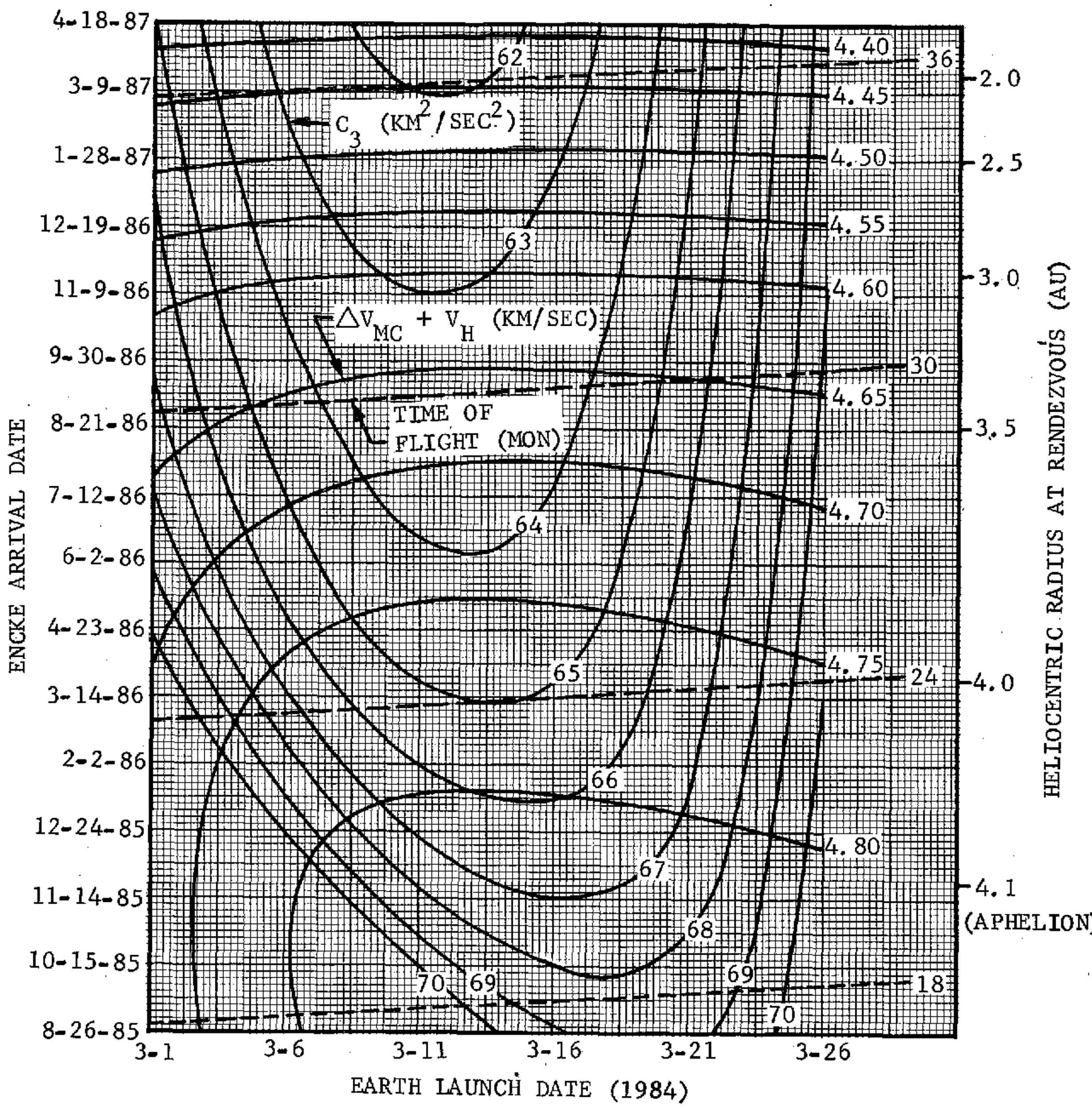

FIGURE VII-2 PERFORMANCE PARAMETERS FOR 1984 LAUNCH/EARLY RENDEZVOUS

A representative case of early rendezvous for the 1984 launch opportunity is depicted in Figure VII-3 and further detailed in Table VII-2. The selected 24-month flight profile is typified by two velocity maneuvers of about the same magnitude which are executed in the vicinity of Encke aphelion. As shown by Figure VII-3, rendezvous occurs shortly after a period of solar interference However, communications could then be maintained without interuption through the perihelion passage.

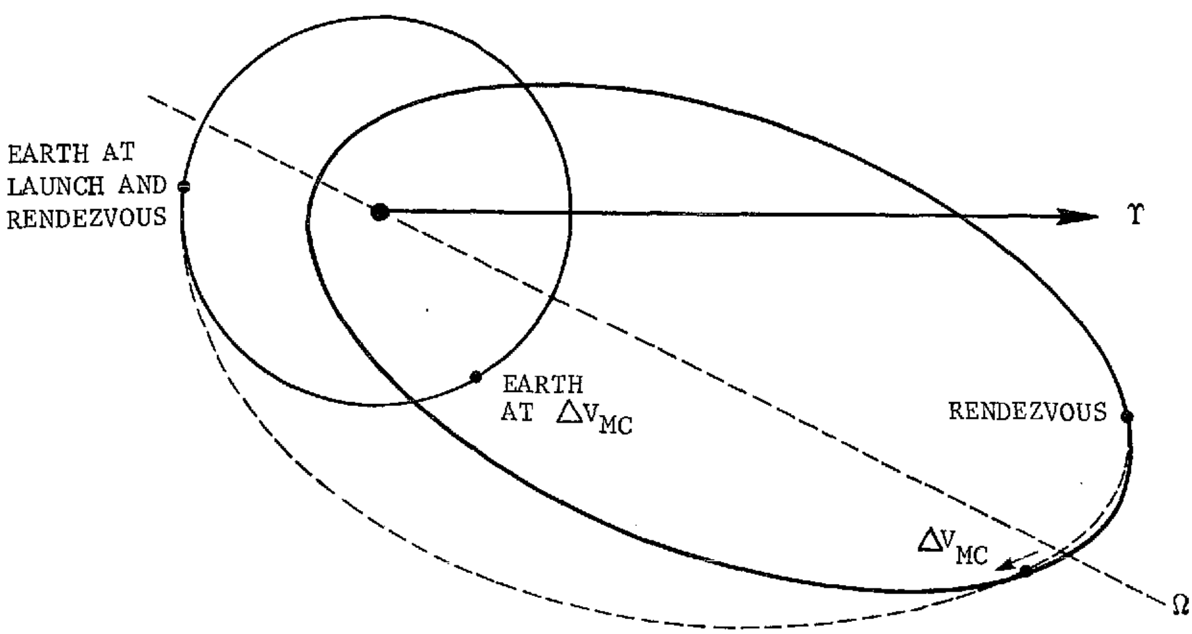

FIGURE VII-3 HELIOCENTRIC GEOMETRY FOR 1984 LAUNCH/24-MONTH RENDEZVOUS

TABLE VII-2 TRAJECTORY PRINTOUT FOR 1984 LAUNCH/24-MONTH RENDEZVOUS

LAUNCH			
LAUNCH DATE	2445773.500 -	3/14/1984. 0. 0. 0	
C3	64.857		
DLA	-22.044		
L-MAN. ANGLE	158.458		
DV1	5.864		
PERIOD	1536.956		
PERIHELION	.994		
APHELION	4.219		
STATE	-.14755195E+09	.18793951E+08	-.32117362E+03
	-.50469782E+01	-.37663916E+02	.18476609E+00
ELEMENTS	.38991287E+09	.61854099E+00	.27858774E+00
	.17276667E+03	-.10025475E+01	.97710347E+00
SPHERICAL	.14874404E+09	-.12371516E-03	.17274122E+03
	.38001010E+02	.27858099E+00	.26236781E+03
MANEUVER			
MANEUVER DATE	2446267.018 -	7/20/1985.12.26.26	
DELTA V	2.887		
DVSUM	8.751		
MAN.-T ANGLE	14.181		
PERIOD	1255.604		
PERIHELION	.524		
APHELION	4.032		
STATE	.50122827E+09	-.27556305E+09	.10223503E+07
	.73163701E+01	.44272396E+01	.11976581E+01
ELEMENTS	.34074409E+09	.77013446E+00	.91308967E+01
	-.29438094E+02	-.16900515E+03	.16965050E+03
SPHERICAL	.57198410E+09	.10240912E+00	.33119908E+03
	.86350511E+01	.79724708E+01	.31178729E+02
TARGET			
ARRIVAL DATE	2446503.500 -	3/14/1986. 0. 0. 0	
VHP	1.897		
DVSUM	10.648		
PHASE	44.051		
SEV	5.908		
(-S)VE	-178.535		
RANGE	4.989		
STATE	.57836276E+09	-.15271831E+09	.24310786E+08
	.22622370E+00	.72629683E+01	.10345063E+01
ELEMENTS	.34074409E+09	.77013446E+00	.91308967E+01
	-.29438094E+02	-.16900515E+03	-.17616851E+03
SPHERICAL	.59867969E+09	.23272687E+01	.34520849E+03
	.73397608E+01	.81025620E+01	.88215953E+02

C. SAMPLE RETURN MISSIONS

The 1987 rendezvous mission was predicted by Figure VII-1 to be insensitive to flight time for the Encke half-orbit preceding the 1990 perihelion passage. This indication has been confirmed by computer analysis. Performance requirements for rendezvous in the vicinity of aphelion display close agreement with the data presented in Section III for encounter in the region of perihelion.

In addition to the inherent opportunities for early science return, the geometry after rendezvous is compatible with return of comet specimens to Earth for laboratory analysis. The opportunity for an Encke sample return mission with total trip time of about 44 months represents an attractive option for comet exploration planning. To implement such a mission, it is expected that the launch capabilities of Shuttle/Centaur or equivalent would be required to accommodate Pioneer-class spacecraft. Moreover, the high relative velocity at Earth approach (about 28 km/sec) would involve re-entry technology comparable to outer planet probes.

A typical flight profile for the 1987 sample return mission is depicted on Figure VII-4. For this example, a 3-month stopover period was assumed. Parametric performance data are presented on Figure VII-5.

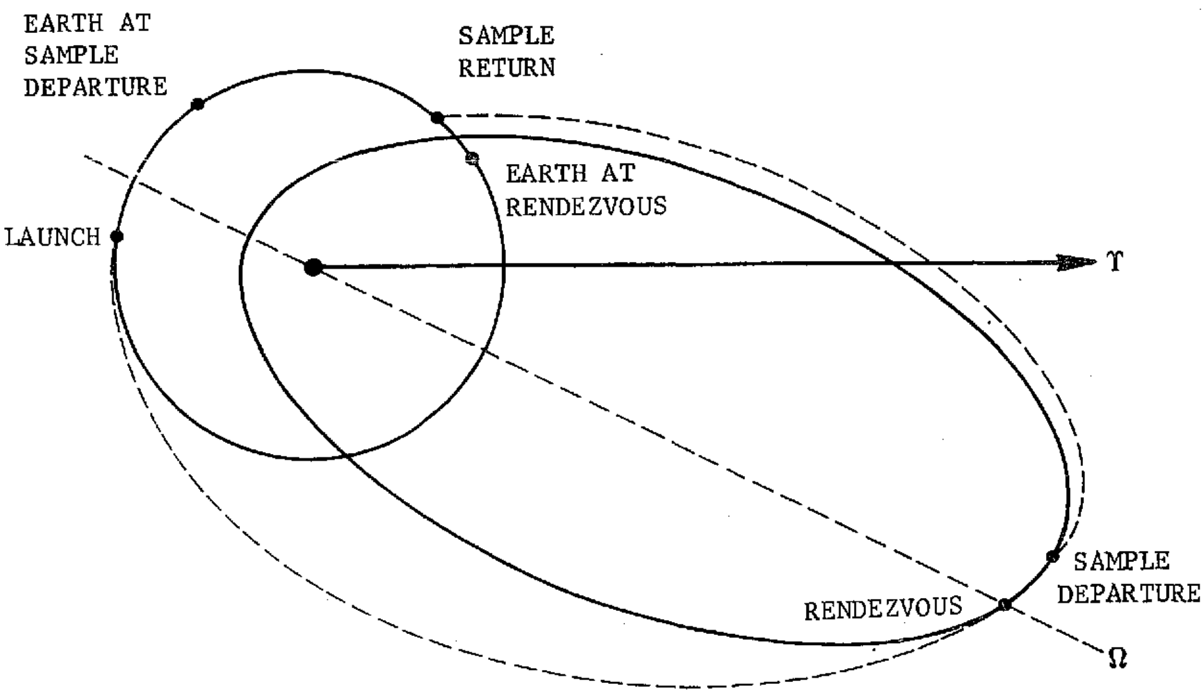

FIGURE VII-4 HELIOCENTRIC GEOMETRY FOR 1987 SAMPLE RETURN

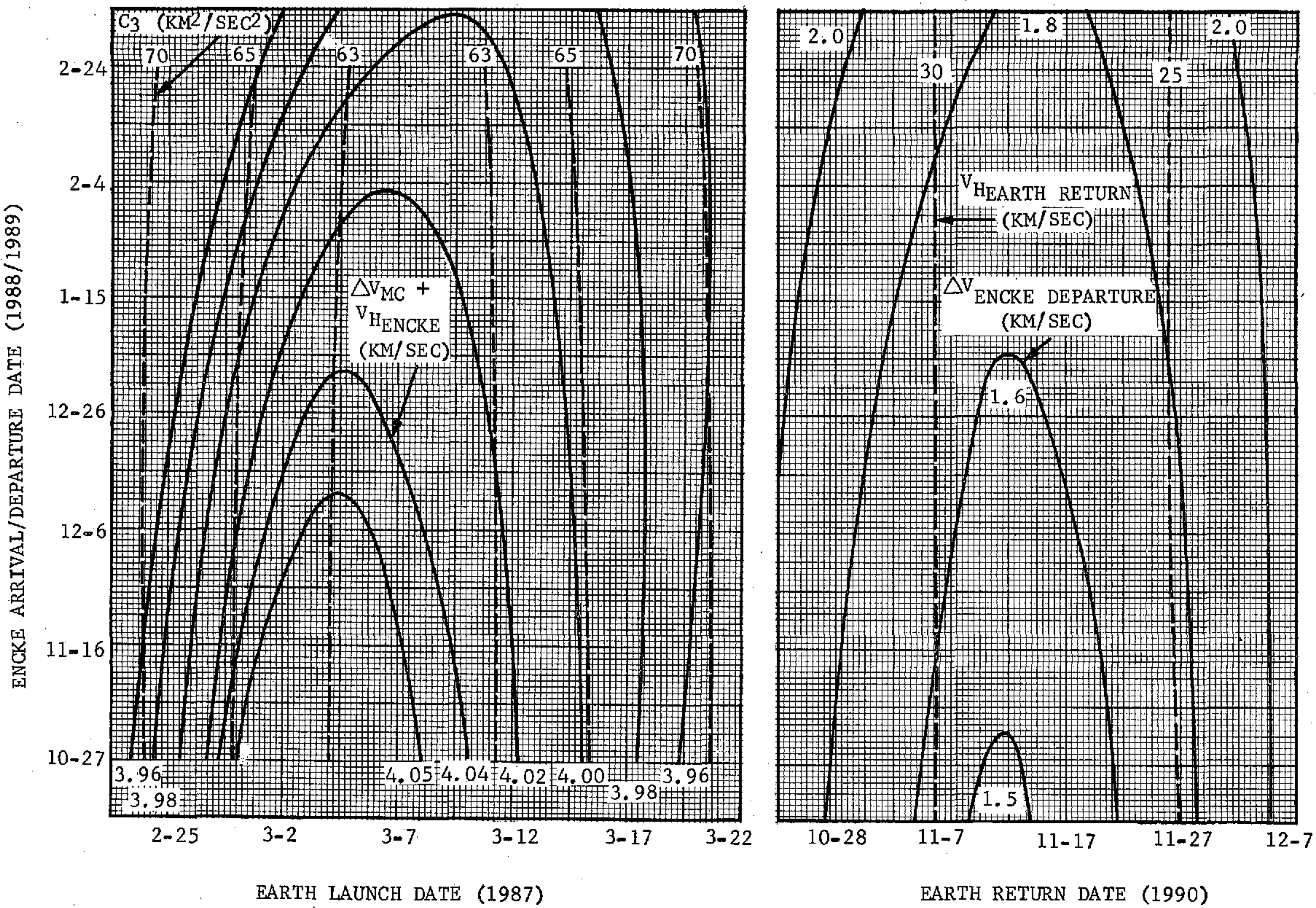

FIGURE VII-5 PERFORMANCE PARAMETERS FOR 1987 SAMPLE RETURN

Analysis of early rendezvous missions launched in 1990 exhibit sensitivity to time-of-flight intermediate to the 1984 and ~~1990~~ 1987 launch cases. Rendezvous performance requirements are in greater conflict with Encke departure requirements than was the case for 1987 launch. However, with availability of Shuttle/Centaur capabilities, the 1990 opportunity could possibly support a sample return mission.

Figure VII-6 illustrates the modified flight geometry necessitated by the 1990 phasing relationships. A stopover of 3 months is depicted. Parametric data are presented on Figure VII-7.

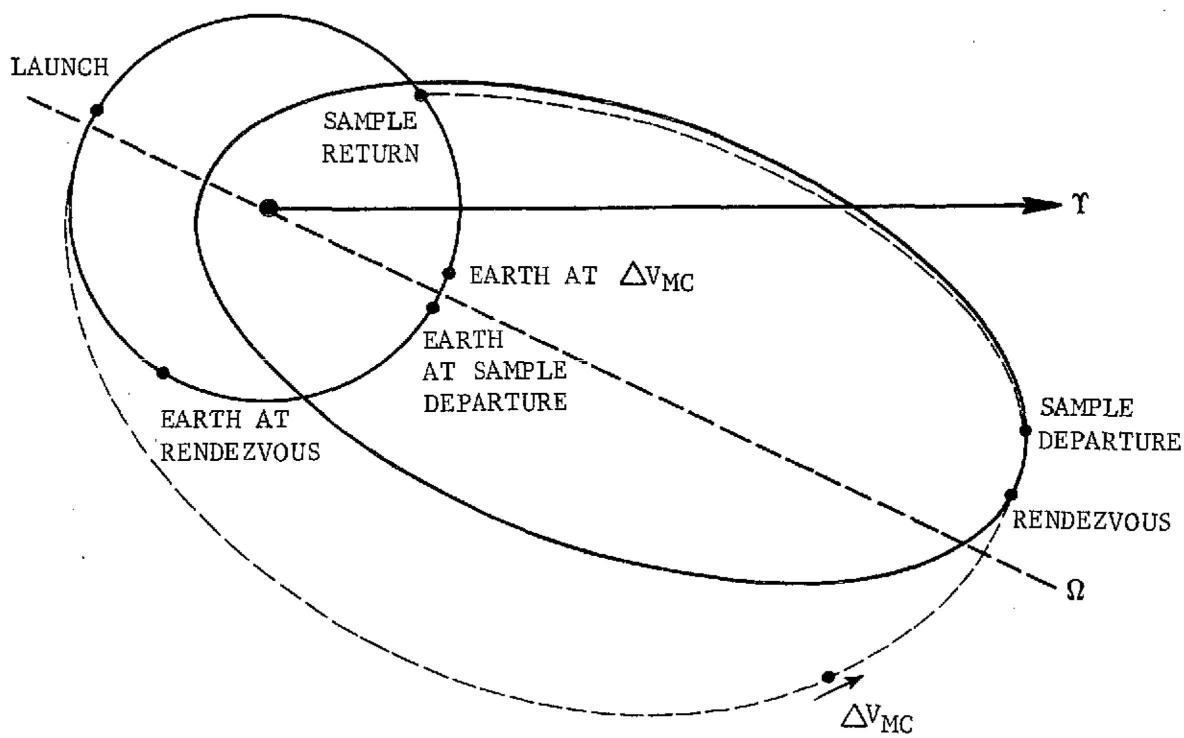

FIGURE VII-6 HELIOCENTRIC GEOMETRY FOR 1990 SAMPLE RETURN

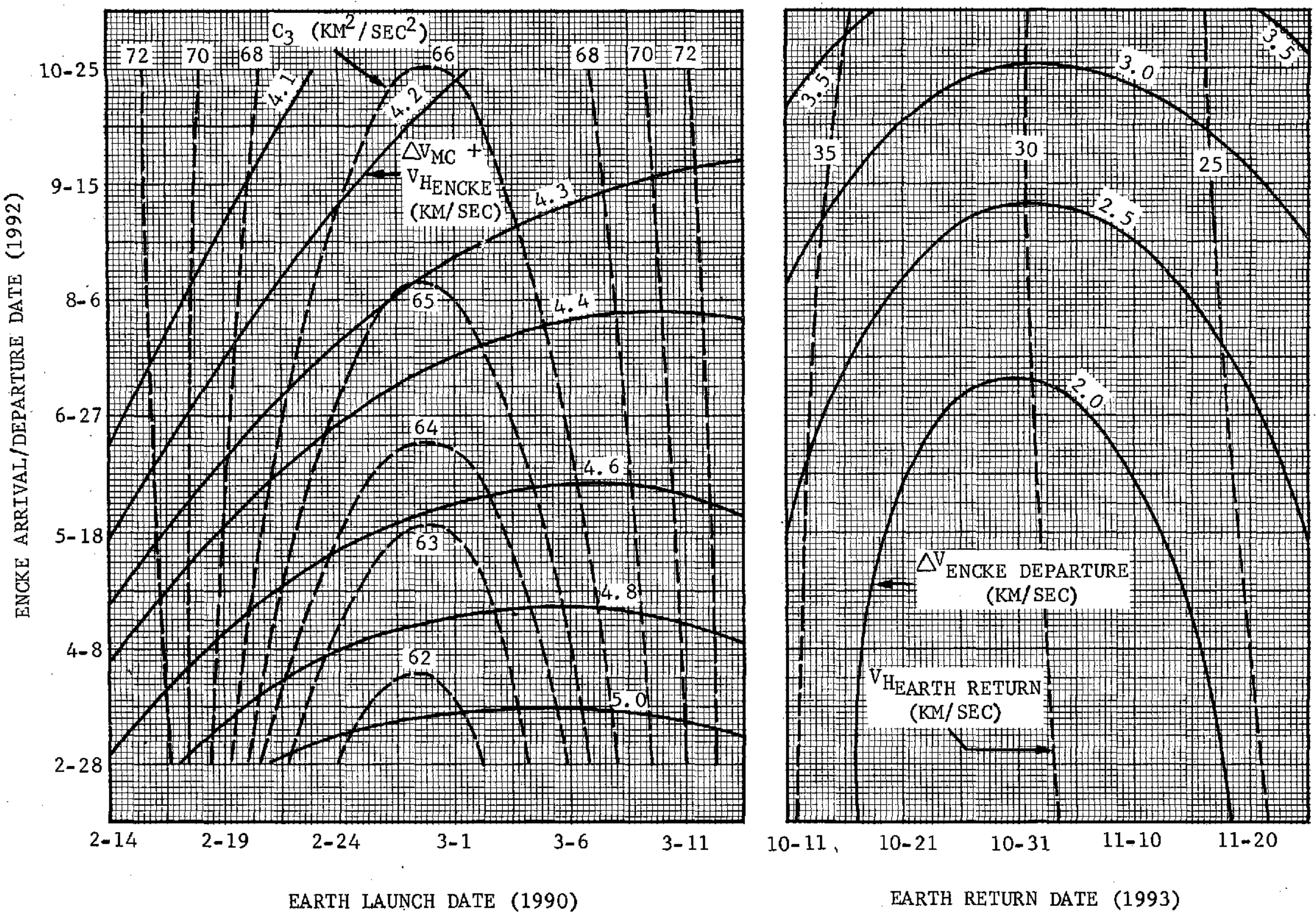

FIGURE VII-7 PERFORMANCE PARAMETERS FOR 1990 SAMPLE RETURN

A summary of performance parameters for the 1987 and 1990 sample return mission opportunities is presented in Table VII-3. Representative spacecraft weights at rendezvous are included on a basis consistent with the conditions specified for Tables I-1 and III-10. The Earth return phase has not been analyzed in quantitative terms but a separate Earth return spacecraft of the Pioneer-class is judged to be appropriate.

TABLE VII-3 REPRESENTATIVE CHARACTERISTICS FOR SAMPLE RETURN MISSIONS

	1987 LAUNCH	1990 LAUNCH
10-Day Earth-Launch Period	3-3 to 3-13	2-19 to 3-1
Encke Arrival Date	10-27-88	5-18-92
Maximum C_3 (KM^2/SEC^2)	63.5	68.0
Maximum ($\Delta V_{MC} + V_H$) (KM/SEC)	4.06	4.64
Midcourse Corrections (KM/SEC)	.15	.15
Net Spacecraft Weight at Rendezvous (KG)		
Titan IIIE/Centaur/TE364-4	340	270
Shuttle/Centaur	520	420
Encke Stopover Interval (DAYS)	90	90
Encke Departure Date	1-25-89	8-16-92
Encke Departure ΔV (KM/SEC)	1.66	2.31
Earth Return Date	11-12-90	10-31-93
Earth Encounter Velocity (KM/SEC)	28.6	30.4
Total Mission Time (MONTHS)	44	44

Detailed data for representative rendezvous and Earth return trajectories for the 1987 and 1990 sample return mission opportunities are presented in Tables VII-4 through VII-7. The print key is contained in Appendix C.

TABLE VII-4 TRAJECTORY PRINTOUT FOR 1987 SAMPLE RETURN (EARTH TO ENCKE)

```
LAUNCH
   LAUNCH DATE      2446862.500 -   3/ 8/1987.  0.  0.  0
   C3                    62.091
   DLA                  -23.079
   L-MAN. ANGLE         166.830
   DV1                    5.762
   PERIOD              1460.411
   PERIHELION              .992
   APHELION               4.046
   STATE           -.14403135E+09    .36051574E+08    -.18339209E+04
                   -.96417885E+01   -.36640902E+02    -.43953471E-01
   ELEMENTS         .37685693E+09    .60609404E+00     .66467822E-01
                   -.14662673E+02    .17878121E+03     .18288438E+01
   SPHERICAL        .14347473E+09   -.70770243E-03     .16594738E+03
                    .37888280E+02   -.66467757E-01     .25525725E+03

MANEUVER
   MANEUVER DATE    2447444.970 -  10/10/1988.11.16.17
   DELTA V                 .407
   DVSUM                  6.169
   MAN.-T ANGLE           1.273
   PERIOD              1422.095
   PERIHELION              .901
   APHELION               4.049
   STATE            .52252972E+09   -.26879956E+09    -.14823395E+06
                    .68070499E+01    .68476238E+01     .14094309E+00
   ELEMENTS         .37023617E+09    .63613808E+00     .88211450E+00
                   -.26283407E+02   -.17015482E+03     .16921593E+03
   SPHERICAL        .58761428E+09   -.14453665E-01     .33277782E+03
                    .96563836E+01    .83631007E+00     .45170250E+02

TARGET
   ARRIVAL DATE     2447461.500 -  10/27/1988.  0.  0.  0
   VHP                    3.662
   DVSUM                  9.831
   PHASE                104.752
   SEV                 -107.357
   (-S)VE               166.121
   RANGE                  3.542
   STATE            .53190340E+09   -.25884371E+09     .53117976E+05
                    .63194901E+01    .70916236E+01     .14098738E+00
   ELEMENTS         .37023617E+09    .63613808E+00     .88211450E+00
                   -.26283407E+02   -.17015482E+03     .17048901E+03
   SPHERICAL        .59154146E+09    .51449239E-02     .33405075E+03
                    .94998398E+01    .85035943E+00     .48295118E+02
```

TABLE VII-5 TRAJECTORY PRINTOUT FOR 1987 SAMPLE RETURN (ENCKE TO EARTH)

```
LAUNCH
   LAUNCH DATE     2447551.500 -   1/25/1989. 0. 0. 0
   C3                    2.748
   DLA
   L-TARG. ANGLE        70.064
   DV1                   1.658
   PERIOD             1224.841
   PERIHELION             .353
   APHELION              4.127
   STATE          .56591478E+09   -.22581252E+09    .94723674E+07
                  .42173659E+01    .46699698E+01    .24081817E-03
   ELEMENTS       .33515553E+09    .84222046E+00    .94901776E+00
                 -.13195225E+03   -.65770829E+02    .17596744E+03
   SPHERICAL      .60937719E+09    .89066106E+00    .33824691E+03
                  .62924394E+01    .21927688E-02    .47915374E+02

TARGET
   ARRIVAL DATE    2448207.500 - 11/12/1990. 0. 0. 0
   VHP                  28.604
   DVSUM                30.262
   PHASE                 0.000
   SEV                   0.000
   (-S)VE                0.000
   RANGE                 0.000
   STATE          .98492015E+08    .11057851E+09   -.11168686E+05
                 -.37022234E+02   -.50636052E+01   -.40001890E+00
   ELEMENTS       .33515553E+09    .84222046E+00    .94901776E+00
                 -.13195225E+03   -.65770829E+02   -.11396826E+03
   SPHERICAL      .14808202E+09   -.43213794E-02    .48308622E+02
                  .37369051E+02   -.61333730E+00    .18778814E+03
```

TABLE VII-6 TRAJECTORY PRINTOUT FOR 1990 SAMPLE RETURN (EARTH TO ENCKE)

LAUNCH			
LAUNCH DATE	2447946.500 -	2/24/1990. 0. 0. 0	
C3	64.057		
DLA	-31.065		
L-MAN. ANGLE	164.739		
DV1	5.834		
PERIOD	1480.295		
PERIHELION	.990		
APHELION	4.094		
STATE	-.13317967E+09	.64655994E+08	-.46843550E+04
	-.16643658E+02	-.34132617E+02	-.13523054E+01
ELEMENTS	.38026994E+09	.61068688E+00	.20395032E+01
	-.25946530E+02	.17978970E+03	.26124329E+00
SPHERICAL	.14804466E+09	-.18129244E-02	.15410438E+03
	.37998363E+02	-.20395025E+01	.24400533E+03
MANEUVER			
MANEUVER DATE	2448491.258 -	8/22/1991.18.11.21	
DELTA V	.505		
DVSUM	6.339		
MAN.-T ANGLE	19.908		
PERIOD	1459.351		
PERIHELION	.928		
APHELION	4.108		
STATE	.43779988E+09	-.38255768E+09	-.54287461E+07
	.91344060E+01	.45034130E+01	.68357166E+00
ELEMENTS	.37667455E+09	.63147842E+00	.44124005E+01
	-.34197240E+02	-.17215650E+03	.16518572E+03
SPHERICAL	.58141945E+09	-.53498168E+00	.31885243E+03
	.10207124E+02	.38399757E+01	.26244059E+02
TARGET			
ARRIVAL DATE	2448760.500 -	5/18/1992. 0. 0. 0	
VHP	4.056		
DVSUM	10.395		
PHASE	122.874		
SEV	64.597		
(-S)VE	-167.059		
RANGE	4.411		
STATE	.56863562E+09	-.22167293E+09	.10513576E+08
	.19778838E+01	.88414923E+01	.65006584E+00
ELEMENTS	.37667455E+09	.63147842E+00	.44124005E+01
	-.34197240E+02	-.17215650E+03	-.17490668E+03
SPHERICAL	.61040634E+09	.98690544E+00	.33870252E+03
	.90833142E+01	.41039973E+01	.77390273E+02

TABLE VII-7 TRAJECTORY PRINTOUT FOR 1990 SAMPLE RETURN (ENCKE TO EARTH)

LAUNCH			
LAUNCH DATE	2448850.500 -	8/16/1992. 0. 0. 0	
C3	5.358		
DLA			
L-TARG. ANGLE	53.715		
DV1	2.315		
PERIOD	1228.178		
PERIHELION	.296		
APHELION	4.192		
STATE	.58028814E+09	-.17985954E+09	.19538746E+08
	-.16429445E+01	.62177070E+01	-.23897592E+00
ELEMENTS	.33576404E+09	.86795937E+00	.22882603E+01
	-.14362400E+03	-.59260676E+02	-.17435790E+03
SPHERICAL	.60783677E+09	.18420744E+01	.34277925E+03
	.64355463E+01	-.21280960E+01	.10480134E+03
TARGET			
ARRIVAL DATE	2449291.500 -	10/31/1993. 0. 0. 0	
VHP	30.443		
DVSUM	32.758		
PHASE	0.000		
SEV	0.000		
(-S)VE	0.000		
RANGE	0.000		
STATE	.11942936E+09	.88283604E+08	-.99700676E+04
	-.37297357E+02	.16593314E+00	-.88924495E+00
ELEMENTS	.33576404E+09	.86795937E+00	.22882603E+01
	-.14362400E+03	-.59260676E+02	-.12064299E+03
SPHERICAL	.14851723E+09	-.38463065E-02	.36472252E+02
	.37308325E+02	-.13657758E+01	.17974510E+03

APPENDIXES

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APPENDIX A - ENCKE EPHEMERIS DATA

Ephemeris data for the 1980, 1984, 1987, and 1990 Encke perihelion passages are presented in Table A-1 and were the basis for the performance data presented in the main body of the report. The 1980 data shown are the same as that used in the earlier phase of this contract, originated from Dr. B. G. Marsden, and was presented in the report "Study of a Comet Rendezvous Mission," Vol. II, TRW Systems Group, 12 May 1972.

The 1984, 1987, and 1990 ephemeris data, were obtained by integrating the comet trajectory forward in time (from the 1980 perihelion) with an n-body trajectory program. The 7 bodies allowed to influence the trajectory were the Sun, Mercury, Venus, Earth, Mars, Jupiter, and Saturn. Nongravitational effects were not included in the simulation, but were considered negligible for purposes of this analysis. The orbit elements were selected from epoch dates close to perihelion to maximize accuracy over the performance data range of comet encounter within 40 days of perihelion. (See Section III)

A significant change occurs in the comet elements between the 1984 and 1987 perihelion passages. This change, causing the period of Encke to reduce by over 7 days, was primarily due to the influence of Jupiter as the comet approached aphelion in late 1985. (Jupiter periodically perturbs the comet elements as the two come close to each other).

TABLE A-1 ENCKE EPHEMERIS DATA

	YEAR OF PASSAGE			
ORBIT ELEMENTS	1980	1984	1987	1990
$T_{Perihelion}$ (Date)	Dec 6.5787	Mar 27.669	July 17.454	Oct 28.467
Semi-Major Axis (10^6 km)	331.8588	331.9709	330.5746	330.4534
Eccentricity	0.846762	0.846335	0.849969	0.850303
Inclination (DEG)	11.9463	11.9276	11.9214	11.9352
Long. of Ascending Node (DEG)	334.2000	334.1869	334.0281	334.0368
Argument of Perihelion (DEG)	185.9784	185.9916	186.2714	186.2534
Period (DAYS)	1206.79	1207.42	1199.82	1199.16
Epoch (Date)	Nov 17.00	Mar 27.54	July 17.54	Oct 25.54

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APPENDIX B - ALTERNATE GRAPHICAL PRESENTATION METHOD

Figures B-1 through B-4 present an alternate graphical presentation method (the contour method) for the 1979 launch/1980 encounter opportunity. Although the ΔV_{Total} parameter is a well behaved function, due to the nature of the problem (two separate conic orbit segments for each trajectory) the components which make up ΔV_{Total} are generally not simple functions.

In this example, C_3 and the midcourse maneuver velocity (ΔV_{MC}) are particularly complex, while the hyperbolic approach velocity (V_H) is reasonably well behaved. For this reason, and also considering that useful information could be obtained by presentation techniques substantially less complex, the method used in Figures III-2 through III-8 was chosen.

$\Delta V_{TOTAL} = \Delta V_1 + \Delta V_{MC} + V_H$ (KM/SEC)

2-24-79 EARTH AT ENCKE DESCENDING NODE

12-6-80 ENCKE AT PERIHELION

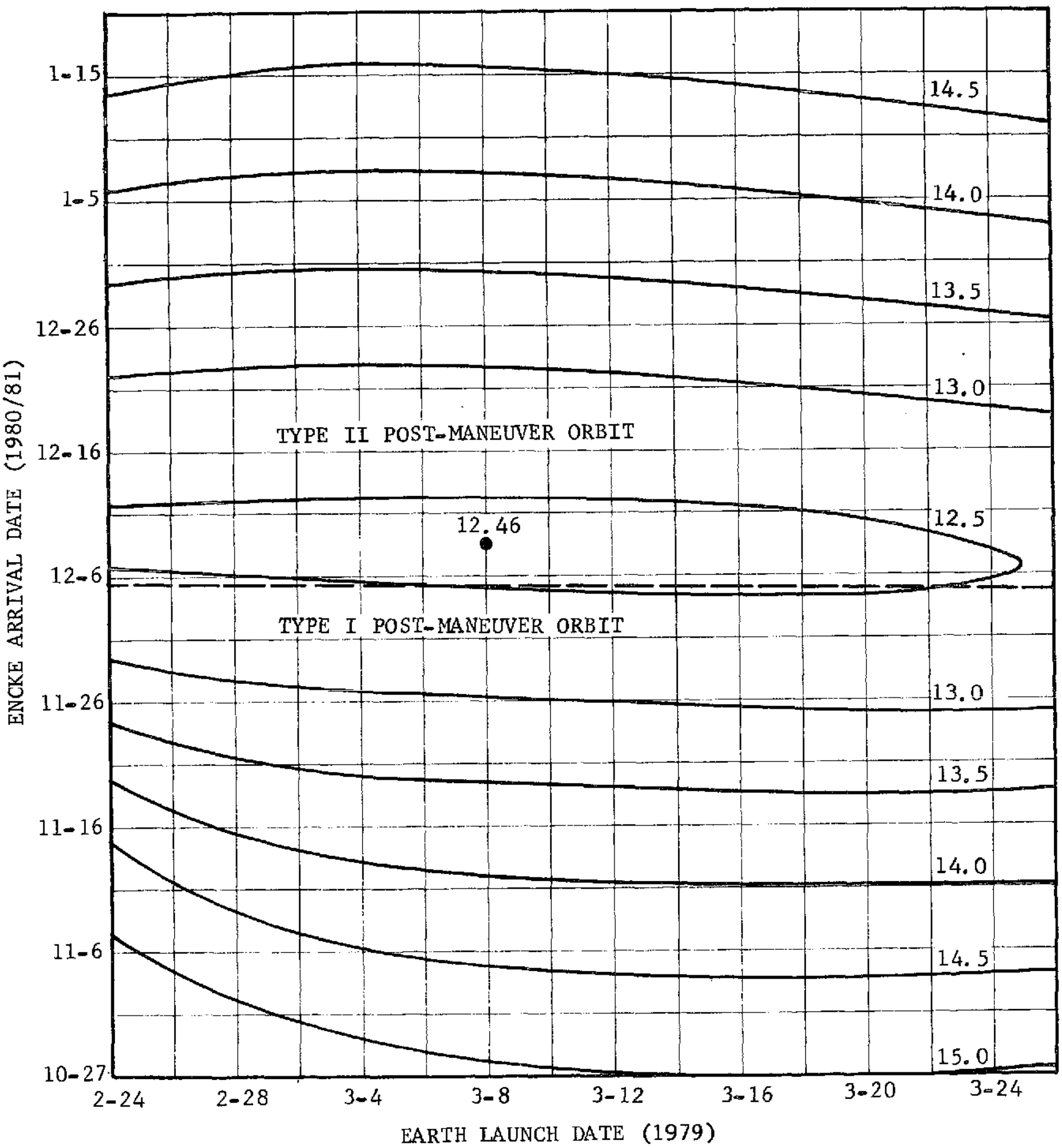

FIGURE B-1 TOTAL VELOCITY REQUIREMENTS FOR 1979/1980 ENCOUNTER

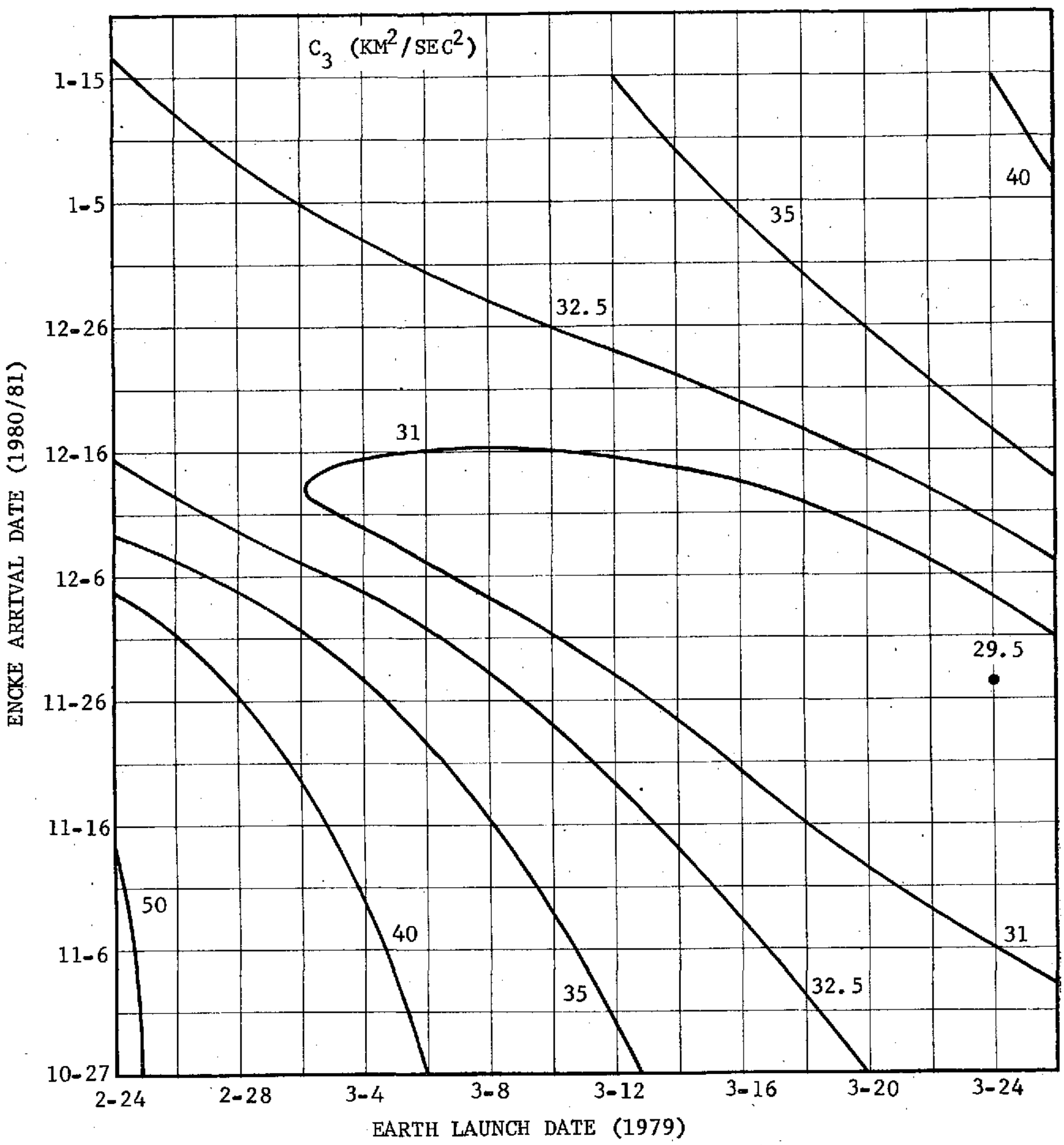

FIGURE B-2 LAUNCH ENERGY REQUIREMENTS FOR 1979 LAUNCH/1980 ENCOUNTER

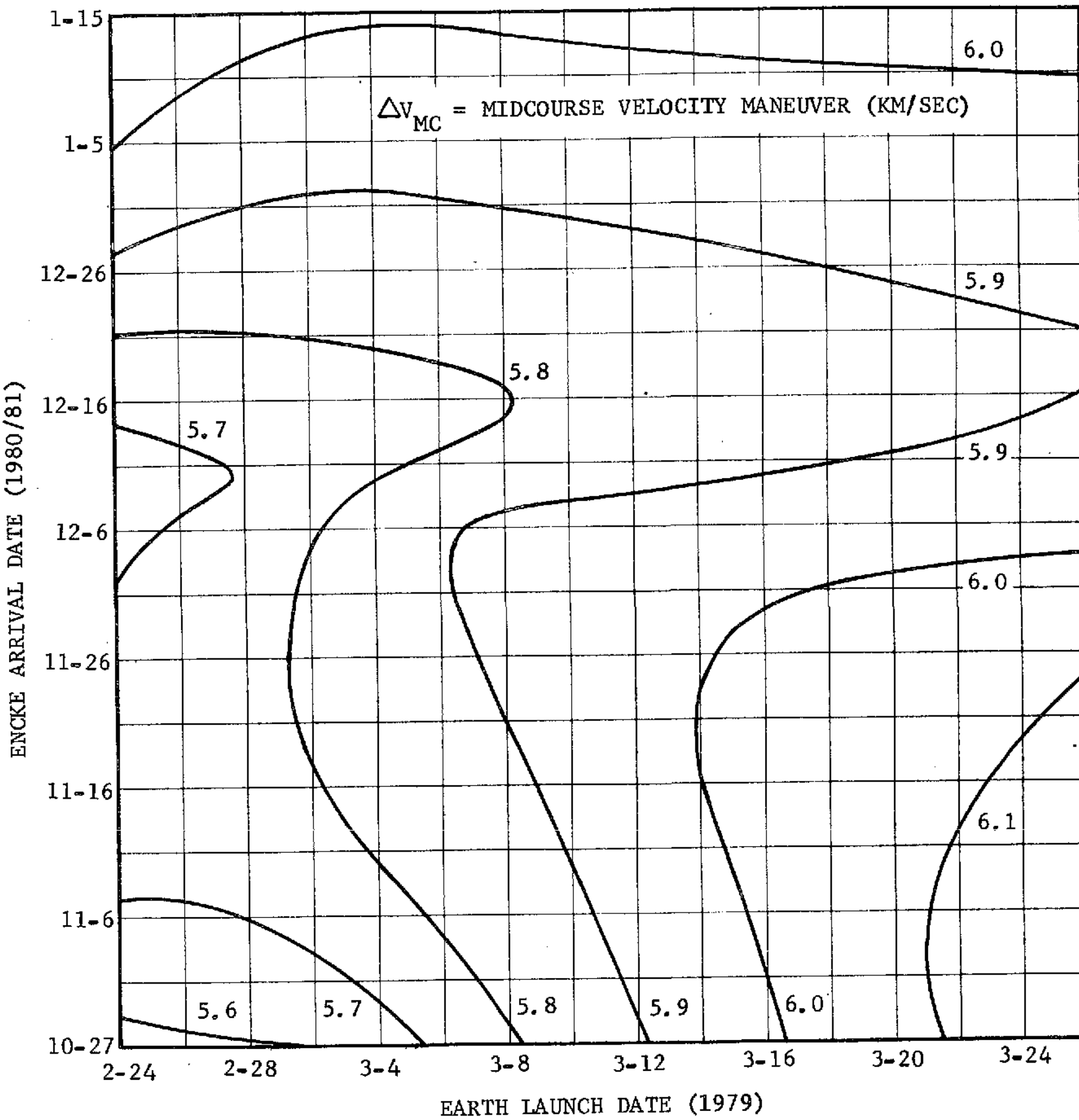

FIGURE B-3 MIDCOURSE MANEUVER REQUIREMENTS FOR 1979 LAUNCH/1980 ENCOUNTER

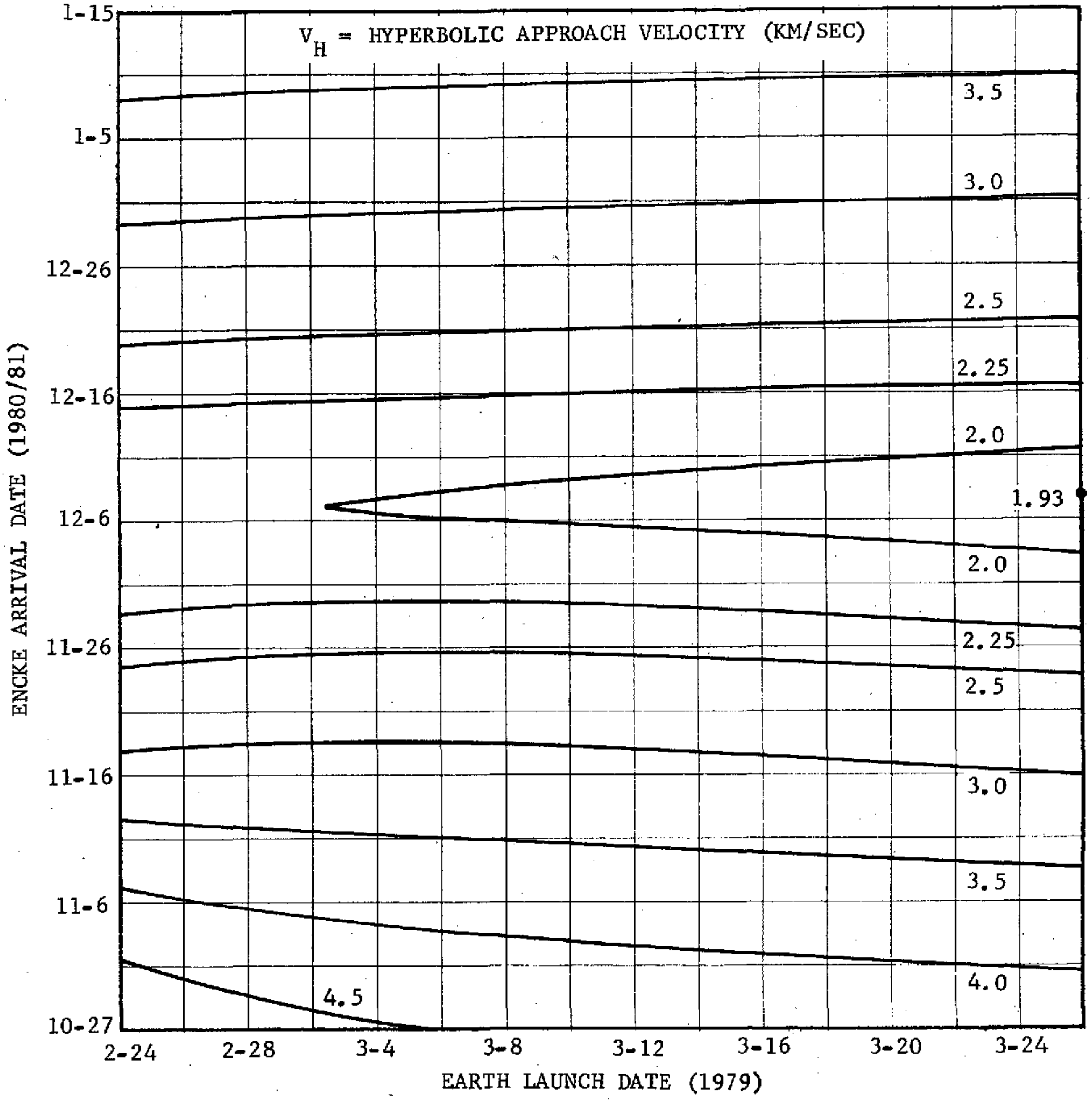

FIGURE B-4 ENCOUNTER VELOCITY CHARACTERISTICS FOR 1979 LAUNCH/1980 ENCOUNTER

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APPENDIX C - TRAJECTORY PRINTOUT INTERPRETATION

All performance data tabulated in Section III were generated with a conic trajectory optimization program called DELOPT (for ΔV optimization). This program was used throughout the parametric performance analysis during the the contract period of work.

Table C-1 defines each parameter in the sample computer printouts tabulated in Tables III-1 through III-8. The normal printout contains a launch block, a maneuver block, and a target block. In the case of the ascending node launch (Table III-1), no maneuver was necessary and the maneuver block is deleted. All units are in kilometers, degrees, and seconds unless otherwise indicated.

TABLE C-1 TRAJECTORY PRINTOUT KEY

LAUNCH

LAUNCH DATE	= Julian date (DAYS) and Calendar date (MO, DA, YR, HR, MIN, SEC) of launch
C_3	= Twice the required launch energy
DLA	= Declination of Launch Asymptote
L-MAN. ANGLE	= Heliocentric Transfer Angle from Launch to Maneuver
or L-TARG. ANGLE	= Heliocentric Transfer Angle from Launch to Target
DV1	= Velocity increment required to leave a 100 nm circular Earth parking orbit to develop specified C_3.
PERIOD	= Orbit period of first (launch to maneuver) orbit segment (DAYS).
PERIHELION	= Radius of perihelion of first orbit segment (AU)
APHELION	= Radius of aphelion of first orbit segment (AU)

The following coordinate systems are Sun-centered and referenced to the ecliptic plane with Aries as the reference direction.

STATE	= Spacecraft state at launch in a right-handed rectangular coordinate system (positive X-axis toward Aries) First Line = X,Y,Z Second Line = $\dot{X},\dot{Y},\dot{Z}$

ELEMENTS = Spacecraft state at launch in terms of the orbital elements of the first orbit segment

First Line = a, e, i

Second Line = Ω, ω, θ

SPHERICAL = Spacecraft state at launch in terms of spherical coordinates.

First Line = Radius, Declination, Rt. Ascension

Second Line = Velocity, Declination of V, Rt. Ascension of V

MANEUVER

MANEUVER DATE = Julian Date (DAYS) and Calendar Date (MO, DA, YR, HR, MIN, SEC) of Maneuver

DELTA V = Magnitude of midcourse velocity maneuver

DVSUM = Sum of DV1 and Delta V

MAN.-T ANGLE = Heliocentric transfer angle from maneuver to target

PERIOD = Orbit period of second (maneuver point to target) orbit segment (DAYS)

PERIHELION = Radius of perihelion of second orbit segment (AU)

APHELION = Radius of aphelion of second orbit segment (AU)

STATE * = Spacecraft state immediately after maneuver (rectangular coordinates)

ELEMENTS * = Spacecraft orbital elements immediately after maneuver

SPHERICAL * = Spacecraft state immediately after maneuver (spherical coordinates)

* The coordinates are the same as defined in the Launch block.

TARGET

ARRIVAL DATE = Julian Date (DAYS) and Calendar Date (MO, DA, YR, HR, MIN, SEC) of Maneuver

VHP = Magnitude of Hyperbolic approach velocity at encounter (before any final maneuver is applied)

DVSUM = ΔV_{TOT} = DV1 + DELTA V + VHP (function being minimized)

PHASE = Approach phase angle (angle between the VHP vector and the comet-position vector at encounter and lies between 0 and +180 deg.)

SEV	= Sun-Earth-Spacecraft angle at encounter. (Angle between the vector from the Earth to the Sun and the vector from the Earth to the spacecraft. Looking downward from the northern ecliptic pole the angle is measured positive clockwise going from the Earth-Sun vector to the Earth - Spacecraft vector and lies between +180 deg. and -180 deg).
(-S)VE	= Anti-Sun-Spacecraft-Earth angle at encounter. (Angle between anti-solar comet tail and the spacecraft-to-Earth vector at encounter. Looking downward from the northern ecliptic pole, the angle is measured positive clockwise going from the spacecraft - Earth vector to the comet-tail vector and lies between +180 deg. and -180 deg.)
RANGE	= Earth-to-Spacecraft range at encounter (AU)
STATE *	= Spacecraft state at encounter. (Rectangular coordinates)
ELEMENTS *	= Spacecraft orbital elements before encounter (Same values as in maneuver block)
SPHERICAL *	= Spacecraft state at encounter (Spherical coordinates)

* The coordinates are the same as defined in the launch block.