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SOME FLIGHT MECHANICS CONSIDERATIONS FOR THE VOYAGER MISSION

By Clyde D. Baker, James C. Blair, and Wilton E. Causey
Aero-Astrodynamics Laboratory

NASA

George C. Marshall Space Flight Center, Huntsville, Alabama

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ABSTRACT

Four aspects of Aero-Astrodynamics Laboratory studies of the Voyager mission are discussed as presented in a status review to Center management. The four areas are (1) the effects of Earth-Mars-Sun geometry on launch opportunities, (2) the influence of trajectory design requirements on the Voyager mission, (3) present performance capabilities of the Saturn V Voyager vehicle, and (4) the proposed launch vehicle load relief control system.

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ASTRODYNAMICS AND GUIDANCE THEORY DIVISION
AERO-ASTRODYNAMICS LABORATORY
RESEARCH AND DEVELOPMENT OPERATIONS

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SUMMARY

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INTRODUCTION

A number of flight mechanics and systems studies for the Voyager mission have been underway in the Aero-Astrodynamics Laboratory. As a review of the status of these studies, several specific topics have been selected for discussion, in order to indicate results, expand on pertinent technical details, and point out problem areas. The discussion is divided into four major categories, as outlined in figure 1. The first section answers the question "Why are there variations in length of launch opportunity for the various mission opportunities?" and presents the advantages and disadvantages of Type II trajectories versus Type I. The discussion of mission constraints in the second section is motivated by the fact that the Voyager trajectory design problem is super-constrained. Several particularly severe constraints are identified as an indication of requirements that must be relaxed. The third section presents current performance data for 1973 and 1975, and indicates how the launch opportunity has shrunk since the time of the original guidelines. Possible alternate mission plans are also discussed. The final section indicates the launch probability increase provided by a proposed load-relief control system, along with functional changes to the Saturn V required by this system.

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*Presented to Center management on August 10, 1967, as Aero-Astrodynamics Laboratory's contribution to the Voyager Status Review.

GENERAL DISCUSSION

1. The Effects of Earth-Mars-Sun Geometry on Launch Opportunities

In this section we will show how the solar system geometry produces a variation in the length of the launch opportunity for the Voyager missions of 1973, 1975, 1977 and 1979. Figure 2, which is a plot of the minimum injection energy, C_3, versus time over the time span of interest for Type I trajectories, illustrates the very sharp minima in injection energy defining each of the launch opportunities, approximately 25 months apart. This 25-month period, known as a synodic period, is defined as the time between the occurrences of a given heliocentric angle between Earth and Mars. Since the launch energy is restricted with the current launch vehicle to between 20 and 30 km^2/sec^2, we are limited to launch during a short span of time for each of these launch opportunities. If the orbits of Mars and Earth were circular, concentric, and in the same plane, then all of these minima would be identical. But the variation in energy required among the launch opportunities, which is apparent in the figure, is caused by the fact that the orbits are neither circular nor coplanar; in fact, Earth and Mars orbits are elliptic, and are inclined to each other by about 1.85 degrees. Next, we will discuss the relative contribution of these two effects to the variation in energy required.

Figure 3 depicts a scale drawing of Earth and Mars orbits showing the true ellipticity of the orbits. It is obvious that the ellipticity is small; the contribution of ellipticity to the variation of energy requirements is also small. This effect accounts for at most a 13 percent variation in the energy required. Also shown in the figure are four typical trajectories for each of the launch opportunities and a line of intersection of the Mars orbit with the ecliptic, which is shown as the line of nodes. What is more significant than the eccentricity effects is the fact that the Mars orbit is inclined to the ecliptic; therefore, the relationship of the planets to the intersection of the two orbital planes for the various opportunities contributes the greater amount to the variation. Figure 4, in which the geometry has been exaggerated for illustration purposes, indicates the effect of the orbital plane inclination for the 1975 transfer. The inclination of 1.85 degrees between the two orbital planes is designated i_M and the inclination of the transfer plane - which is defined by the three points: Earth at launch, Mars at arrival, and the sun - is designated i_T. It can be seen that in 1975 i_T is greater than i_M; in fact, i_T is in the vicinity of 4 or 5 degrees for 1975. If we have a large value of i_T, the energy requirements go up sharply when we consider having to significantly modify the initial velocity imparted to the spacecraft by the Earth's orbital velocity in the ecliptic of 30 km/sec. Therefore, it is the inclination of the two orbital planes and the position of the planets with respect to the line of nodes that is the most significant factor affecting the variation in launch energy required.

The question has been raised, Why not go to Type II trajectories and get a very long launch window? Figure 5 is an attempt to illustrate some of the factors involved and some of the energy considerations for a Type II trajectory. In the lower right-hand corner of the figure, we have shown schematically the definition of Type I and Type II trajectories. The smaller circle represents Earth's orbit and the larger circle represents Mars' orbit. A Type I trajectory is defined as one which travels a heliocentric angle of less than 180 degrees, and a Type II trajectory is defined as one which travels a heliocentric angle of greater than 180 degrees. The Type II trajectory is longer, requires longer travel time, longer communication distances, etc. The plot in figure 5 of minimum C_3 versus launch date for the 1975 opportunity is an expansion of the Type I curve of figure 2 in the 1975 region. We have also shown a minimum energy requirement for the Type II trajectory. Constraints on launch opportunity other than energy exist, but what is indicated here is the restriction imposed on launch opportunity from energy considerations only. We are restricted in launch energy under current Voyager guidelines to a C_3 value of 23.9 km^2/sec^2, as indicated in the figure, so that launch cannot be achieved for energy values greater than this. However, there are two sides of the energy question, one being the launch energy as indicated by C_3 and the second being the energy required for deceleration into orbit at Mars. This second restriction is indicated on the chart as a restriction that the V_∞ (which is hyperbolic excess velocity at Mars) be not greater than 3.4 km/sec. This value of V_∞ would permit a terminal orbit of 1100 x 10,000 km altitude, with no rotation of its line of apsides. Considering both energy requirements, we are allowed a launch opportunity (from energy considerations alone) that spans the time shown by the solid lines for the two types of trajectories. The Type I trajectory opportunity is restricted to about 23 days, and the Type II trajectory opportunity to about 55 days. This 22-day increase that the Type II shows over the Type I would have to be bought at the expense of some other considerations. For example, the travel time is from 40 to 80 percent longer for the Type II. Communication distance is longer on the order of 30 to 40 percent, yielding a lower data rate. The Type II is more sensitive to guidance errors so that an additional midcourse could be required. Also, the communication geometry would be different from the Type I so that different articulation would very likely be required for the antenna system.

2. The Influence of Trajectory Design Requirements on the Voyager Mission

There are a great many constraints and design requirements imposed on the Voyager trajectory by science and engineering considerations of the launch vehicle, spacecraft, and capsule. Figure 6 is a partial listing of the known constraints that have been imposed on the Voyager mission, many of which are conflicting. Because there is no trajectory that will satisfy all of these constraints, there is no solution for the Voyager mission design problem if we must operate under all of these conditions. However, a positive approach can be taken which attempts to identify from this list a small number of constraints which are particularly limiting from a mission design standpoint - a set of constraints which must be modified or eliminated if we are to achieve a Voyager mission. Figure 7 is such a list. First, there was originally a requirement that Canopus not be occulted for the first 30 days in Martian orbit. This constraint has now been relaxed to indicate a maximum occultation of Canopus of one and one-half hours per orbital revolution. Canopus occultation is a particularly restrictive constraint from a trajectory design viewpoint as will be shown later. Second, the requirement for a standard Saturn V launch vehicle, implying a standard Saturn V control system, imposes an unrealistic limitation on launch probability, and this will be the subject of discussion in the fourth major item of the presentation. The third constraint listed in figure 6 is an indication of conflicting science and engineering requirements on terminal orbit design. Requirement 3a, which indicates that there be one hour's direct link communication from the capsule to earth following landing of the capsule, imposes a restriction on the elevation of the earth angle from the capsule so as to avoid undesirable effects of the Martian atmosphere on the communication signal. Requirement 3b specifies that the orbit of the spacecraft be positioned so as to provide good lighting for the orbital photography experiments. The requirements 3a and 3b are conflicting constraints; that is, there is not one orbit that satisfies both of these requirements at the times under consideration. A possible solution to this problem would be to provide the capability for a large orbit change maneuver. Then the vehicle could go into an orbit suitable for 3a, deploy the capsule and maneuver to an orbit favorable for photography, satisfying the 3b constraint. However, this would require additional propellant, possibly additional tank design and another burn of the spacecraft engine.

The fourth item of figure 6 is not so much in the nature of a constraint, but is in the nature of some information that needs to be identified at an early point in the Voyager program. We need to identify just what the magnitude and velocity of all of the ΔV separation increments are which are imposed on the shroud pieces and other debris following injection into the interplanetary transfer. We need to establish the velocity increments imposed by venting also, in order to define the effect of this on the quarantine requirement that we not contaminate Mars. The fifth constraint is the requirement that we have two standard spacecraft. If our performance under the guidelines does not provide sufficient launch probability, we may have to consider compromising the mission to go with one spacecraft (or modifications thereof) in order to achieve a mission during the launch opportunity.

Returning to the first constraint, the next two figures will indicate some of the conflicting considerations that are imposed by occultation requirements. Figure 8 depicts a Mars arrival geometry for a typical 1973 mission. In this picture Mars is viewed from the earth so that the earth vector is perpendicular to the plane of the figure. The sun vector is in the direction indicated, and the vector to Canopus is nearly mutually perpendicular to the earth and sun vectors. Thus, it can be seen that a requirement that the terminal orbit not occult Canopus would very likely conflict with the requirement for not occulting the sun. Figure 9 is a chart which is used in terminal orbit design. The abscissa is the inclination of the orbit to the Martian equator and the ordinate is the orientation of the line of apsides with respect to the direction of asymptote of the approach trajectory, so that for a given orbit these two quantities completely specify the orientation of that orbit with respect to Mars. The gray area is the area in which the sun is occulted, and there is a requirement that this area must be avoided. The red area is an area in which the earth is occulted, and there is currently a requirement that we must have earth occultation in order to perform atmospheric density profile experiments. It might be mentioned parenthetically that the sub-satellite experiment described by SSL would eliminate the requirement for earth occultation and would be favorable from a trajectory design standpoint. The blue area, which accounts for the major portion of this plot, is the region in which Canopus is occulted. It is obvious that the Canopus occultation is the most restrictive occultation requirement. There are other constraints that are not directly indicated but which can be superimposed on the terminal orbit design chart, such as the satellite photography lighting considerations and the capsule direct link communication requirement which were mentioned earlier. Also there is a limited ΔV capability on the vehicle which limits the deviation from the ψ = 120-degree line. In summary, the trajectory design problem is super-constrained. While the list of constraints to be relaxed is not unique, it is an indication of what must be modified in order to achieve a Voyager mission.

3. Present Performance Capabilities of the Saturn V/Voyager Vehicle

This section shows the current Saturn V/Voyager performance for the 1973 and 1975 missions. Figure 10 is a plot of Mars arrival date versus earth departure date for the 1973 Type I trajectory mission. It might be well to review some of the constraints imposed on the launch opportunity length at this point. The opportunity is restricted on the left by a requirement that the declination of the launch asymptote should not exceed 36 degrees. This constraint is associated with the restrictions of launch azimuth from the Cape. On the right the launch opportunity is constrained by injection energy, C_3. The bottom restriction is due to the limited ΔV budget and its effect on the hyperbolic excess velocity at Mars (V_∞); and the top restriction is due to decreasing opportunity length.

The outer solid lines are associated with the original VPE-14 guidelines which existed when we assumed project responsibility from JPL, and the dashed lines reflect our current guidelines imposed by the Voyager project manager (setting a given weight and ΔV budget). Notice that the current guidelines have resulted in the loss of several days of launch opportunity compared with the original VPE-14 guidelines. Currently, for the 1973 mission the launch opportunity is around 40 days maximum, decreasing to 25 days for the later arrival dates. This is probably not unreasonably short; however, in 1975 the situation worsens considerably.

Figure 11 is a similar plot for the 1975, Type I mission and here we can see that the current guidelines, including a growth from a 5000-pound capsule to a 7000-pound capsule, indicate that we have a launch opportunity of only 13 to 15 days. While there has not yet been a comprehensive launch probability analysis run, it has been JPL's experience that a launch opportunity of less than about 20 days will not permit a suitably high probability of launch during that time. We are currently below that 20-day value for the 1975 mission. Although there are some things that can be done to increase the length of our launch opportunity, they result primarily in very small increases while requiring increased operational complexity. If we come near the closing of a launch opportunity without achieving launch, we may have to consider some alternate mission or contingency plan. A number of these alternatives have been indicated on figure 12. Here, we consider three orders of mission compromise and indicate what <u>gross</u> increase in launch opportunity these compromises would allow. The final column on figure 12 is the gross increase in launch opportunity. This does not take into account the fact that taking the action indicated in the second column would require delay on the pad. For instance, considering Mode 1-a, the action of off-loading of propellant in the spacecraft and removal of one capsule prior to launch would require a period of time on the pad which would have to be subtracted from the 14-day gross increase indicated in the last column. Of all of the possibilities listed on the chart (and there are possibilities other than these six) that can be considered, the one requiring the least operational complexity is Mode 3, where the launch is made with standard vehicle, two spacecraft on board, and nominal configuration. Then, instead of injecting both spacecraft into interplanetary transfer, the first spacecraft would be separated and left in the earth orbit, and the mission would be continued with only one lander and one orbiter. This would provide a gross increase of approximately 19 days in the launch opportunity, and in this instance, there would be no operational delay to subtract from it. This is not intended to be a comprehensive survey of alternate plans, but is simply an indication of what could be done in the way of mission compromise in order to achieve a mission during a given launch opportunity.

4. Proposed Launch Vehicle Load Relief Control System

Aero-Astrodynamics Laboratory has been studying the possibility of modifying the flight control of the first stage Saturn V/Voyager vehicle in order to enable the vehicle to withstand higher winds without requiring a structural beef-up. Figure 13 is a plot of the probability of not experiencing a wind speed which exceeds the structural design limit versus the four launch opportunities. The launch opportunity points have been connected by lines to show the trends. The data on this chart are based on Apollo-type design considerations (wind profiles and anomalies) for the control system. The lower curve indicates the probability that results from using the current Apollo-type attitude control system for the Voyager vehicle, i.e., a control system which consists of attitude and attitude rate feedbacks only. The upper band indicates the improved probability that we currently predict will result from using a load-relief control system currently under study by Honeywell, Inc. The load-relief system is represented by a band instead of a curve because there are some unresolved questions that are still under study on that system. The lower edge of the band may be considered a pessimistic estimate of what the load-relief system will buy, and the upper line of the band could be considered an optimistic estimate of load relief system performance. Using the Apollo-type control system in 1973 produces nearly 99 percent probability, whereas by the 1979 launch opportunity, this value has dropped to 50 percent because of the variation in expected wind value as the month of the launch opportunity changes from year to year. Use of the load-relief system should result in an increase in 1979 probability to between 80 and 90 percent.

The message from this chart is: If we take the Saturn-Apollo design limit of a probability of 95 percent as a guideline, then in 1973 we would have no problem with our current Apollo-type attitude control system. However, in the later years, it becomes imperative that we have some sort of relief on the launch probability problem, and we would definitely recommend the load-relief system for these later opportunities. Although other load-relieving systems are under study, the Honeywell system has received the most attention. Figure 14 provides a listing of the functional changes to the Saturn V control system which are required by this load-relief system. First, we would need to add an accelerometer in the pitch channel and one in the yaw channel, with their associated filters and gain scheduling. Also, it may be necessary to provide some continuous gain scheduling in the pitch and yaw sensor channels, which would not necessarily be a sophisticated time program but could be ramped gains. The system is still under study and we are seeking simpler methods to obtain the same amount of load relief. However, the functional changes that we have listed here need to be translated into implementation requirements so that a realistic assessment of the reliability decrement provided by the added hardware may be determined.

CONCLUSIONS

1. For the Mars mission, the most significant factor producing a variation in length of the various launch opportunities is inclination of Mars' orbital plane to the ecliptic.

2. Type II trajectories can provide somewhat longer launch opportunities, but are undesirable because they require longer trip time, longer communication distance, and higher guidance sensitivity.

3. The trajectory design problem is super-constrained by the many science and engineering requirements of the capsule, spacecraft, and launch vehicle. In order to achieve a mission, some of the constraints must be modified or eliminated.

4. The launch opportunity length has been reduced several days by current guidelines, compared with original VPE-14 guidelines. The 1975 Type I launch opportunity is only 13 to 15 days long, which is likely to be too short to allow a high probability of launch.

5. Additional days of launch opportunity can be achieved toward the end of an opportunity by off-loading or removing various payload segments, thus compromising the basic two-planetary-vehicle mission.

6. It is necessary to provide a load-relief control system for first stage flight in order to achieve a reasonable launch probability for the 1977 and 1979 missions without modifying the Saturn V structure. A 1973 mission would not require a load-relief system.

7. The hardware and reliability implications of a proposed load-relief control system should be assessed. The reliability information can then be input to launch probability studies to determine the requirement for the system in the 1975 mission.

I. THE EFFECTS OF EARTH-MARS-SUN GEOMETRY ON LAUNCH WINDOWS.

II. THE INFLUENCE OF TRAJECTORY DESIGN REQUIREMENTS ON THE VOYAGER MISSION.

   A. LAUNCH PHASE ENGINEERING

   B. SPACECRAFT ENGINEERING

   C. SPACECRAFT SCIENCE

   D. CAPSULE BUS ENGINEERING

   E. CAPSULE SCIENCE

III. PRESENT PERFORMANCE CAPABILITIES OF THE SATURN V/VOYAGER VEHICLE.

IV. PROPOSED LAUNCH VEHICLE LOAD RELIEF SYSTEM.

FIGURE 1. OUTLINE OF AERO-ASTRODYNAMICS PRESENTATION

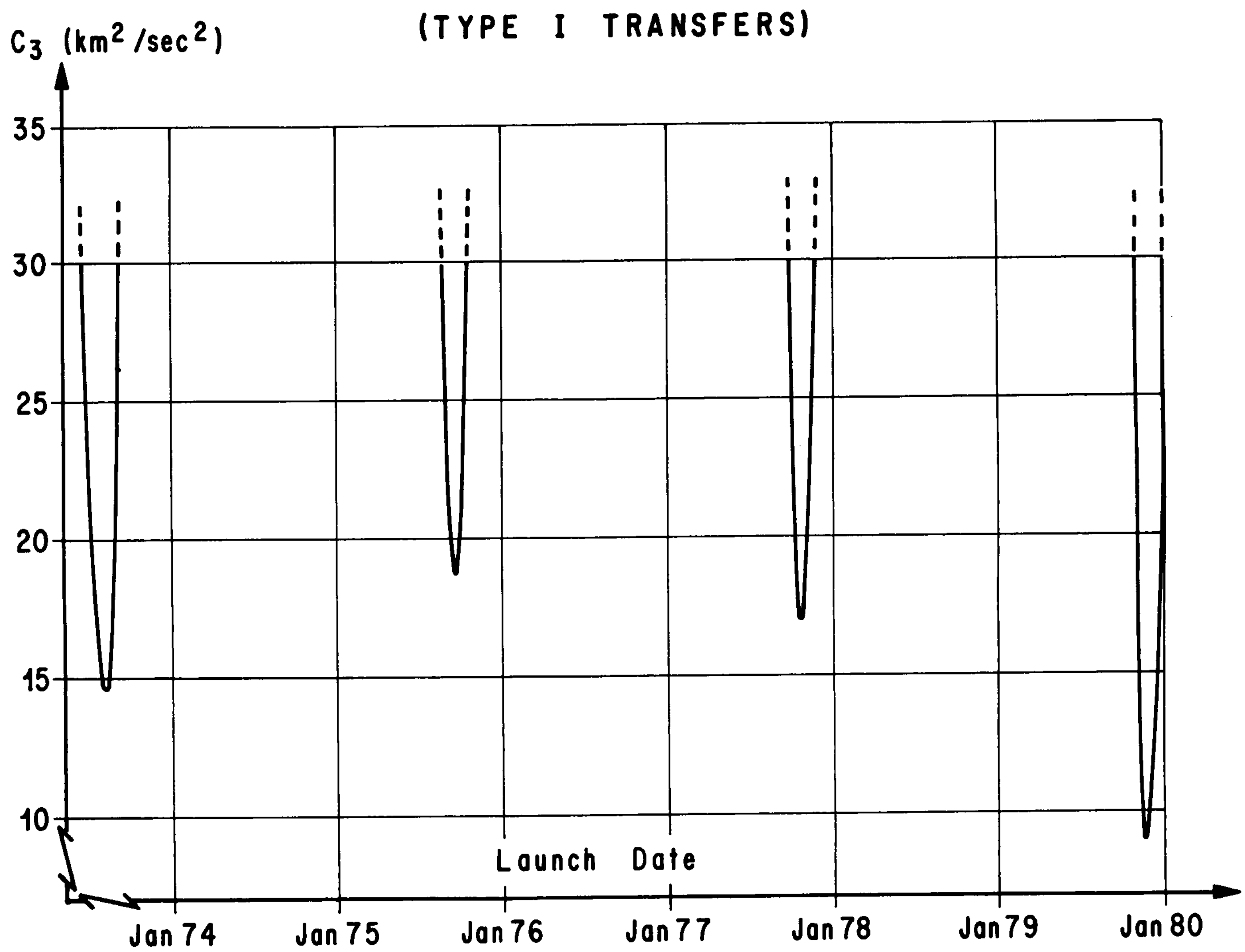
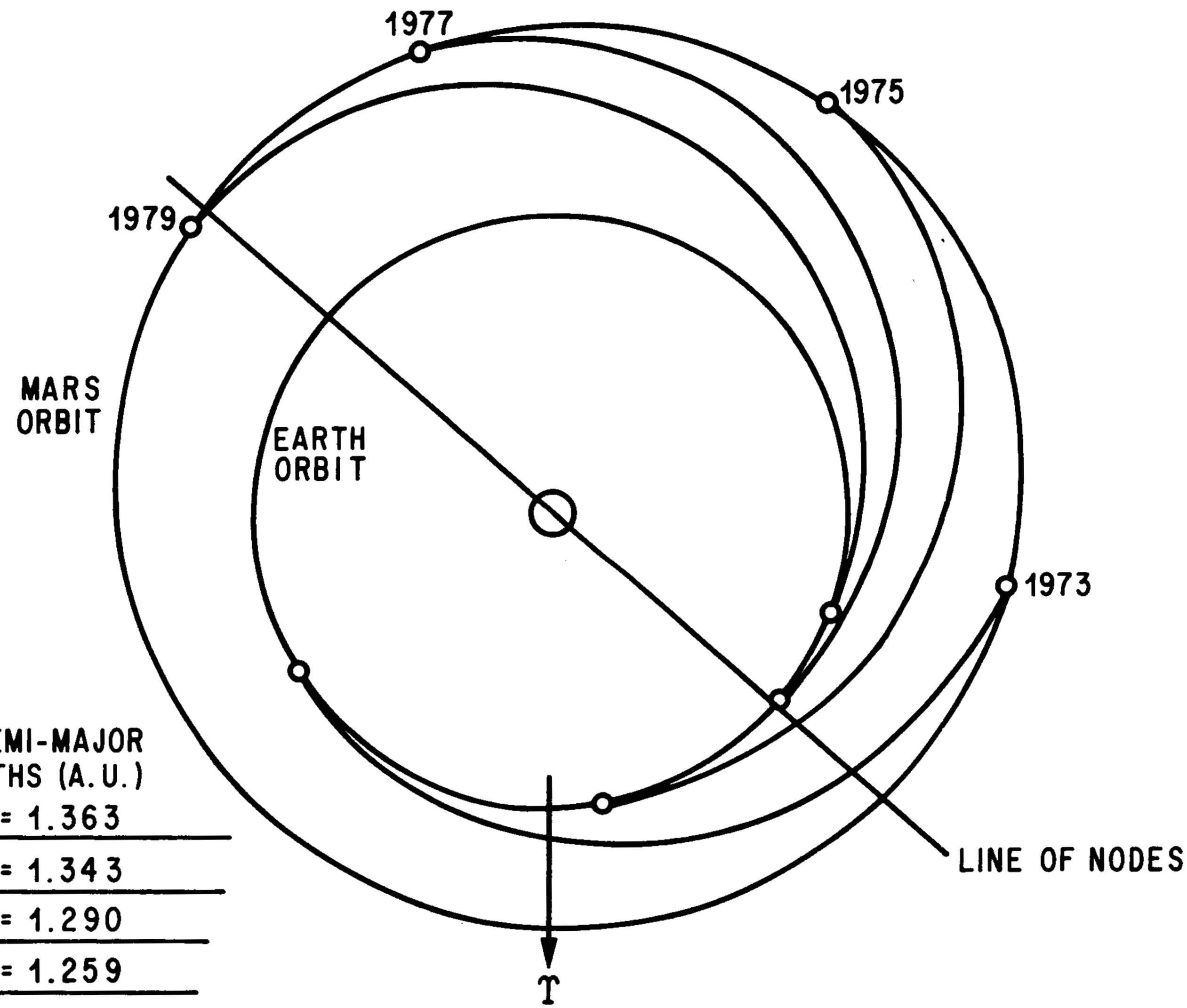


FIGURE 2. LAUNCH ENERGY REQUIREMENTS



RELATIVE SEMI-MAJOR AXIS LENGTHS (A.U.)

1973	a = 1.363
1975	a = 1.343
1977	a = 1.290
1979	a = 1.259

FIGURE 3. TYPICAL MARS MISSION TRAJECTORY PROFILES

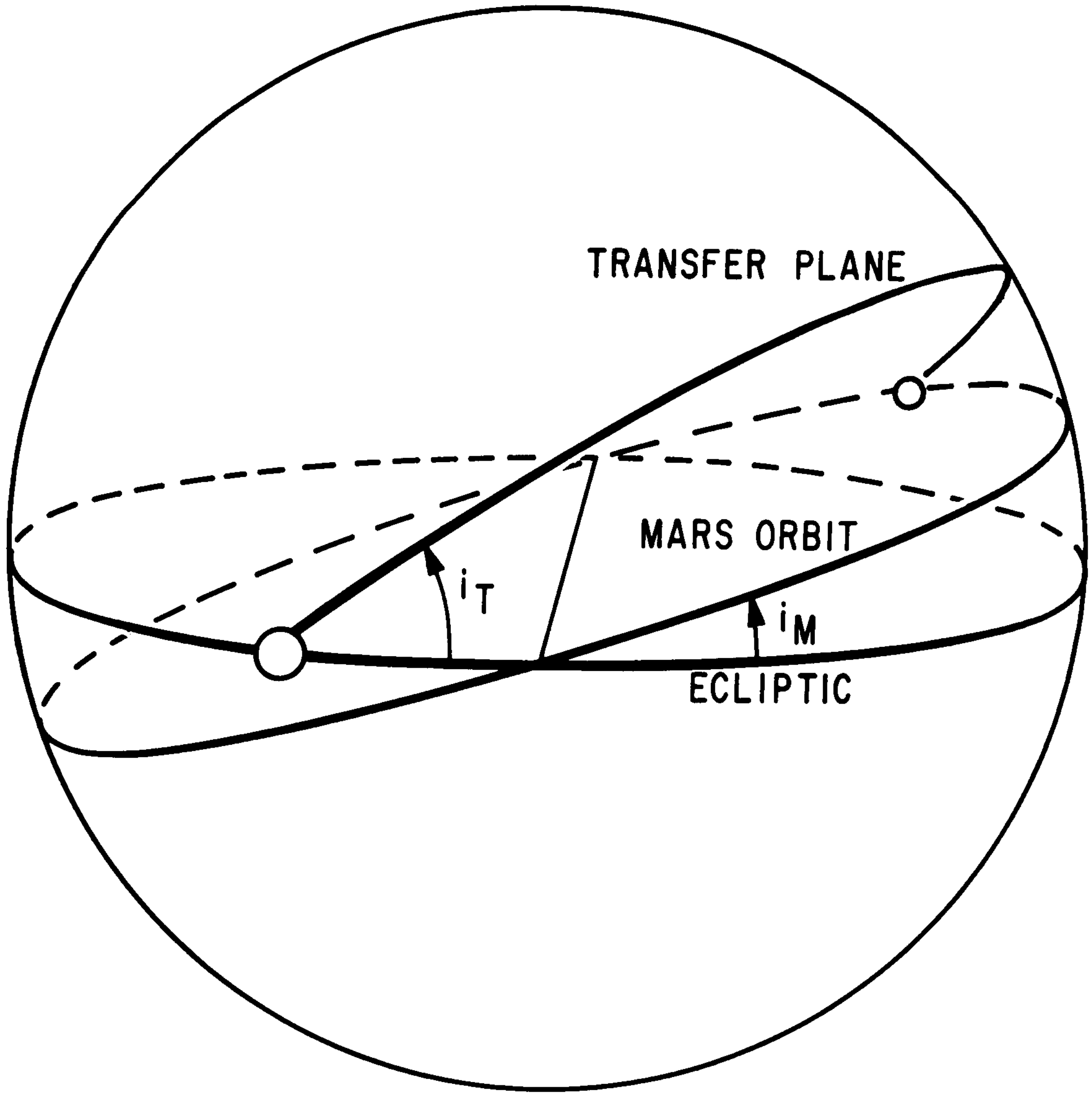


FIGURE 4. EXAGGERATED GEOMETRY OF THE 1975 TRANSFER SHOWING $i_T > i_M$

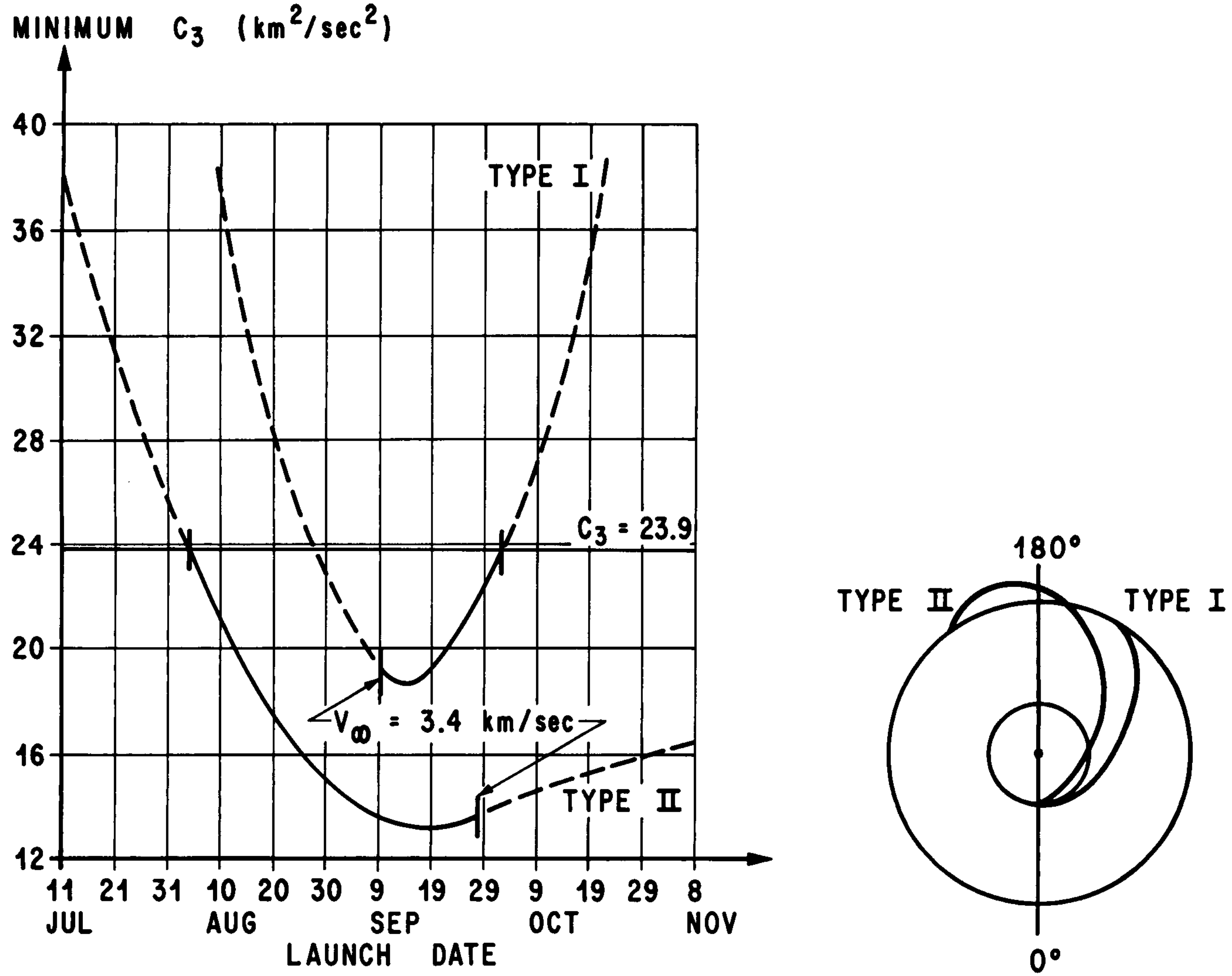


FIGURE 5. ENERGY RESTRICTIONS 1975 MISSION

Launch Vehicle

1. Standard Saturn V
2. Restriction on S-IVB restart and maximum park orbit coast time
3. Launch azimuth 45° to 115°
4. Minimum Launch Window -- 1 hour
5. Maximum C_3 = 23.9 km^2/sec^2 at 115° launch azimuth
6. Quarantine Bias Requirements on Debris at Injection/Separation

Launch Opportunity

7. Launch period: 30 days minimum (1973) for other years commensurate with specified launch probability.
8. Arrival date separation: 8 days (1973), 4 days (1975, 1977, 1979)
9. Type I trajectories preferred.
10. DLA > 5°, < 51°
11. ΔV capability = 1.95 (1973), 1.92 (1975)
12. Capsule Weight 6000#-73; 7000#-75
13. Standard spacecraft
14. Constant arrival date
15. Two spacecraft
16. Planetary vehicle weight = 24,200 lb.

Satellite Orbit Selection

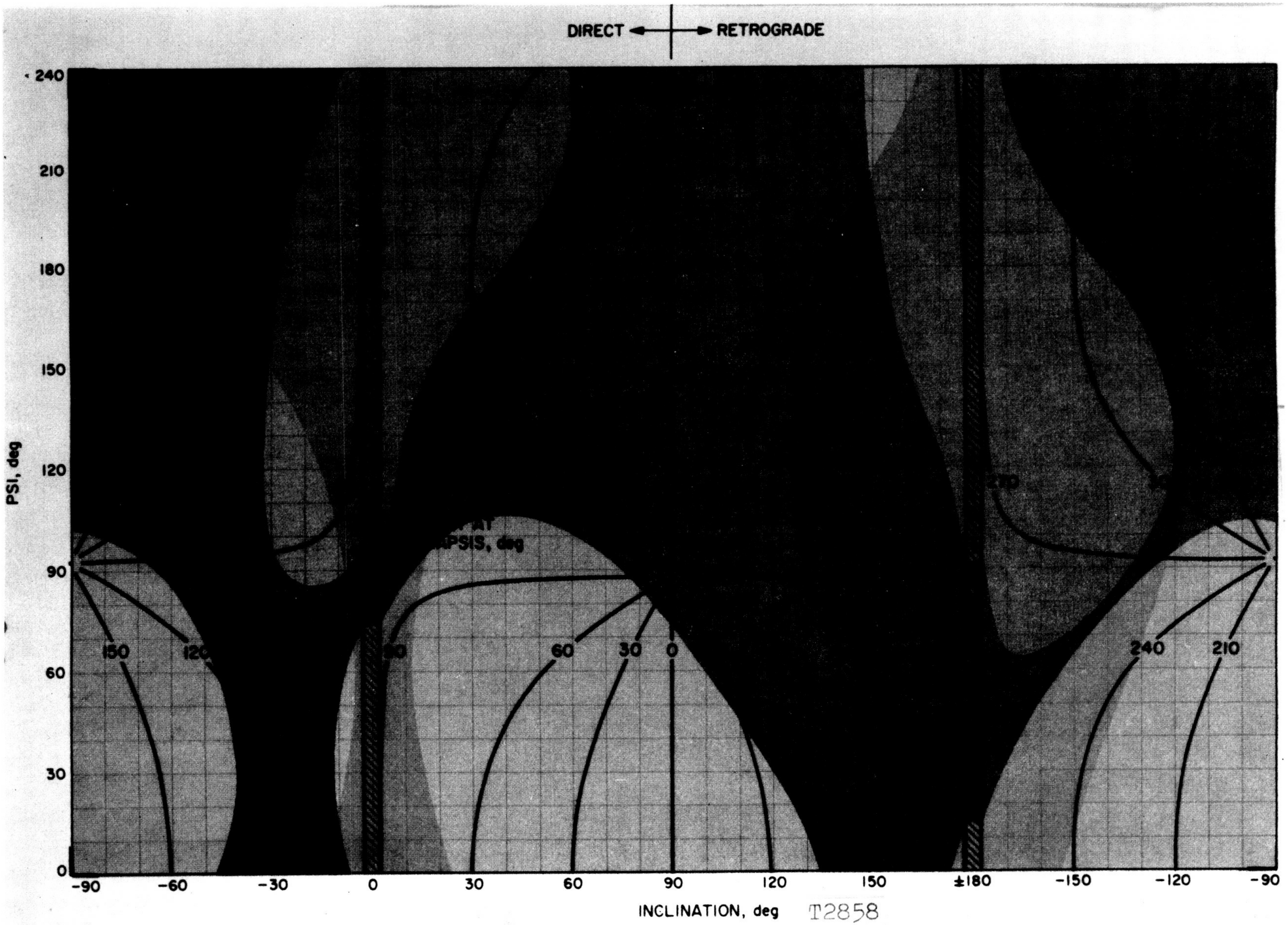
17. Apsidal rotation: 20° capability
18. Capsule landing sites 10°N to 40°S
19. Maintain fixed-antenna communication to 30 days after arrival.
20. Maximum sun occultation after 30 days in orbit: 8% of orbital period or 60 minutes.
21. No loss of Canopus lock for first 30 days
22. Maximum loss of Canopus lock < 1.5 hours
23. Angle between ⊥ to orbit plane and Mars-Earth line > 5°
24. Orbit inclination > 30°
25. Latitude of sub-periapsis point -- 60°S to 40°N
26. First three months, angle between orbit plane and terminator > 30°
27. Second three monehts, angle between orbit plane and terminator < 30° for maximum of one month
28. Capsule deorbit 3-12 days after insertion (max. of 30 days)
29. Capsule impact 15° to 30° from terminator
30. No sun occultation for first 30 days
31. Earth occultation time limited by data return rate
32. First three months, angle from sub-periapsis point to terminator > 0°, < 45°
33. Second three months, angle between sub-periapsis point and terminator > -30°, < 90°
34. Earth occultation line covers wide range of latitude and solar zenith angles
35. Periapsis altitude -- 500 to 1500 km
36. Apoapsis altitude -- 10,000 to 20,000 km
37. Desirability to look at same area of planet again after a period of time
38. Angle between orbital plane and the ecliptic < 45°
39. Communication requirement of 3 hours view time from earth before propulsive maneuver
40. Capsule direct link communication for at least 1 hour after landing

FIGURE 6. VOYAGER CONSTRAINTS

1. CANOPUS OCCULTATION
2. STANDARD SATURN CONTROL SYSTEM
3. (a) CAPSULE DIRECT LINK COMMUNICATION
   (b) LIGHTING CONSTRAINT ON SATELLITE ORBIT
4. QUARANTINE BIAS REQUIREMENTS ON DEBRIS AT INJECTION/SEPARATION
5. TWO SPACECRAFT

FIGURE 7. SOME CONSTRAINTS REQUIRING MODIFICATION OR ELIMINATION TO ACHIEVE A PRACTICAL VOYAGER MISSION

FIGURE 8. VOYAGER ENCOUNTER GEOMETRY



PERIAPSIS ALTITUDE: 1000 km
APOAPSIS ALTITUDE: 20000 km
LAUNCH DATE: AUGUST 17, 1973
ARRIVAL DATE: MARCH 1, 1974

INCLINATION LIMITATION
SUN OCCULTATION
EARTH OCCULTATION
CANOPUS OCCULTATION

FIGURE 9. TERMINAL ORBIT DESIGN CHART

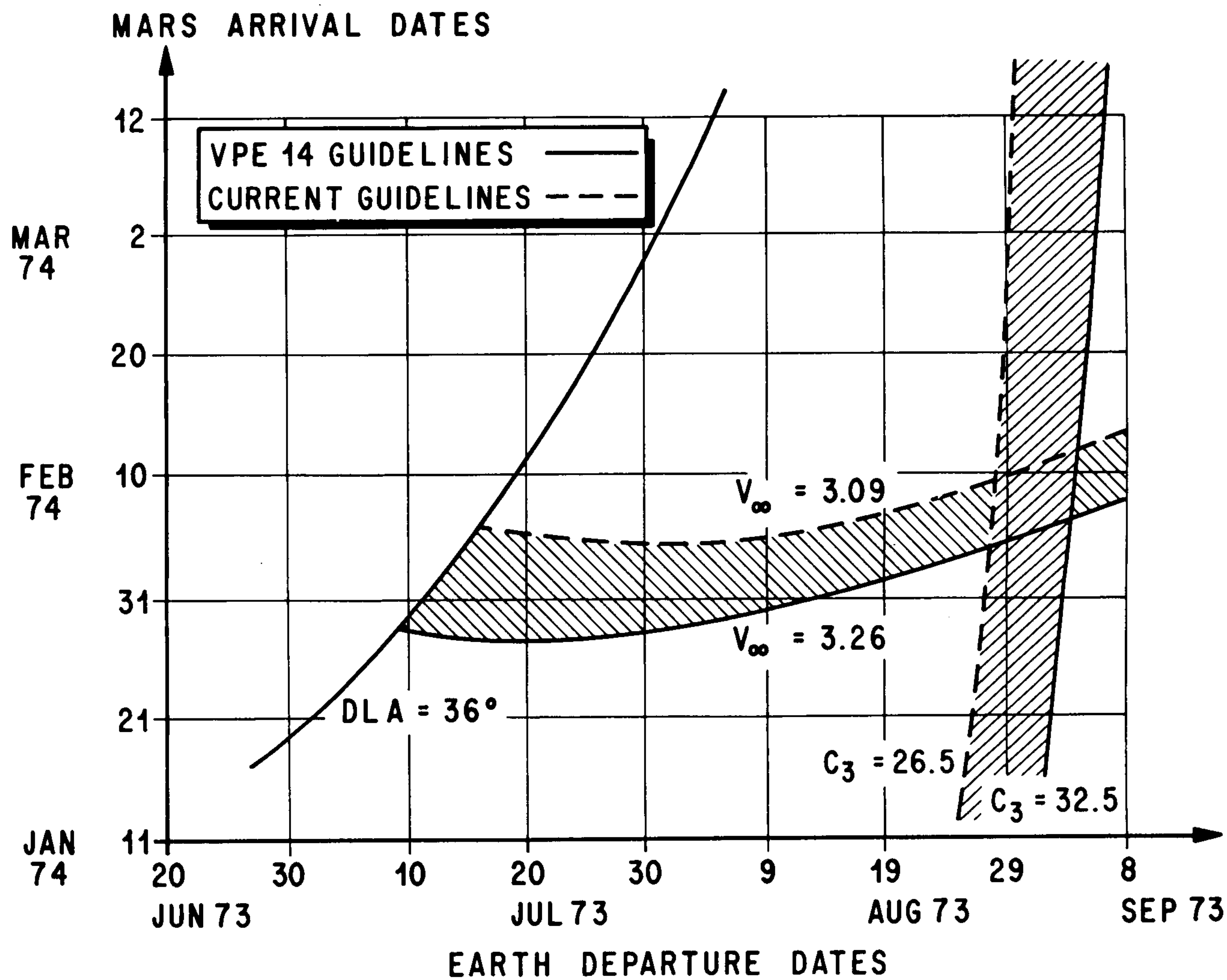


FIGURE 10. EARTH LAUNCH OPPORTUNITIES FOR THE 1973 MISSION

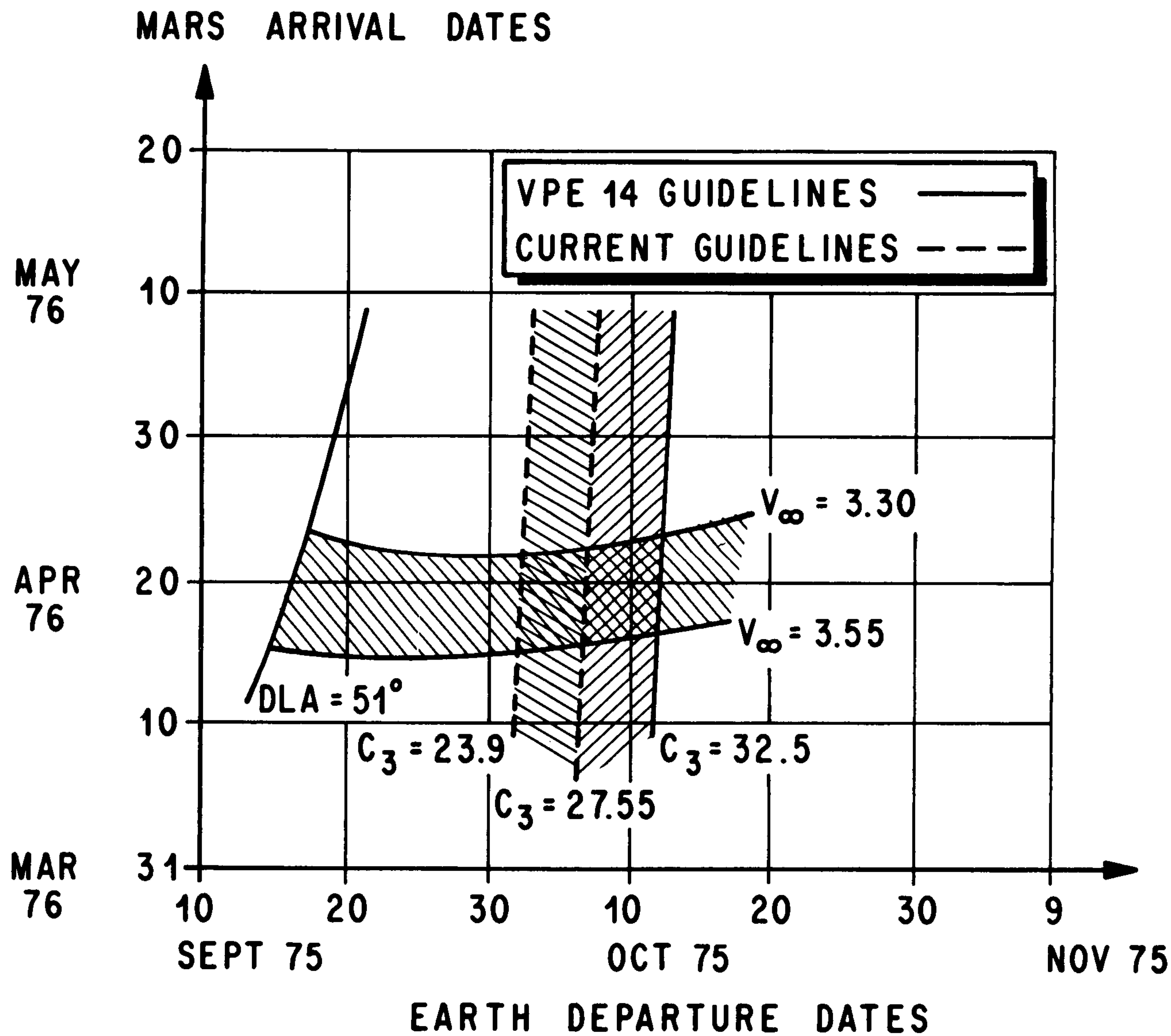


FIGURE 11. EARTH LAUNCH OPPORTUNITIES FOR THE 1975 MARS MISSION

(BASELINE: 1975 LAUNCH PERIOD ≈ 15 days)

MODE	ACTION	MISSION COMPROMISE	INJECTION C_3 (km^2/sec^2)	GROSS INCREASE IN LAUNCH OPPORTUNITY* (days)
1	OFF - LOAD PROPELLANT IN 1 S/C, SEPARATE CBS PRIOR TO MARS ORBIT	1 LANDER 2 ORBITERS	30.5	7
1A	OFF - LOAD PROPELLANT IN 1 S/C, REMOVE 1 CBS BEFORE LAUNCH	1 LANDER 2 ORBITERS	37.5	14
2	OFF - LOAD PROPELLANT IN BOTH S/C, SEPARATE BOTH CBS PRIOR TO MARS ORBIT	NO LANDER 2 ORBITERS	37.5	14
2A	OFF - LOAD PROPELLANT IN BOTH S/C, REMOVE BOTH CBS BEFORE LAUNCH	NO LANDER 2 ORBITERS	55	25
3	DROP 1 P/V IN EARTH ORBIT	1 LANDER 1 ORBITER	45	19
3A	REMOVE 1 P/V BEFORE LAUNCH	1 LANDER 1 ORBITER	52	23

* OPERATIONAL DELAY ON PAD NOT CONSIDERED

FIGURE 12. CONTINGENCY PLAN

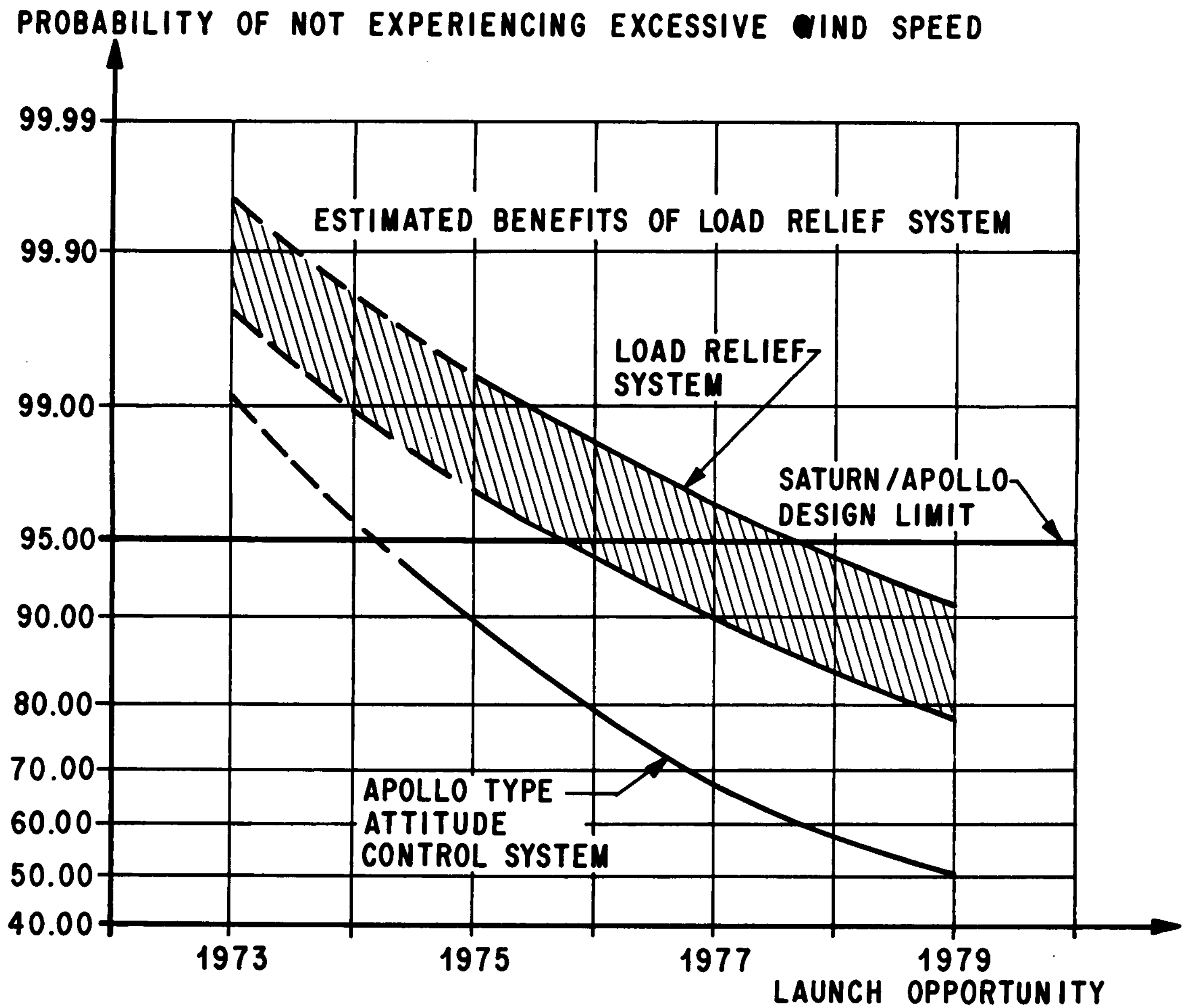


FIGURE 13. 45-FOOT CYLINDRICAL SECTION, SYNTHETIC WIND PROFILES

- Add two body-mounted accelerometers with associated signal channels (filters, gain scheduling);
- Provide continuous gain scheduling on pitch and yaw sensor channels (may be able to use some discrete gains).

FIGURE 14. FUNCTIONAL CHANGES TO SATURN V REQUIRED BY LOAD RELIEF SYSTEM

NASA TM X-53654 APPROVAL

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The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense of Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This report has also been reviewed and approved for technical accuracy.

E. D. Geissler
Director, Aero-Astrodynamics Laboratory

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