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PAYLOAD CAPABILITIES OF SATURN IB-CENTAUR FOR LAUNCH OPPORTUNITIES TO MARS IN 1971, 1973, AND 1975 AND TO VENUS IN 1972 AND 1973

By Gary D. Sagerman and Tim J. Kreiter

Lewis Research Center
Cleveland, Ohio

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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PAYLOAD CAPABILITIES OF SATURN IB-CENTAUR FOR LAUNCH OPPORTUNITIES TO MARS IN 1971, 1973, AND 1975 AND TO VENUS IN 1972 AND 1973

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SUMMARY

A study was made to determine energy requirements for interplanetary missions to Mars and Venus. Corresponding payloads using the Saturn IB-Centaur launch vehicle were also determined. The 1971, 1973, and 1975 Mars launch opportunities and the 1972 and 1973 Venus launch opportunities were investigated. Three mission profiles were considered: a flyby, an orbiter, and a combined orbiter-lander. A retrostage using Earth-storable propellants was provided for the orbiter and orbiter-lander missions, which require retropropulsion near the target planet. Data presented for flyby missions are also applicable for a direct lander mission which uses only atmospheric braking.

Trajectories for the flyby mission were selected on the basis of minimum daily injection energy throughout the launch opportunity. For orbiter and orbiter-lander missions, the orbit insertion maneuver must be considered, and trajectories were therefore selected on the basis of the minimum sum of the hyperbolic velocity at Earth departure and the hyperbolic velocity at arrival at the target planet. This results in very nearly maximum payloads in orbit. Both type I trajectories (heliocentric travel angle less than 180°) and type II trajectories (heliocentric travel angle greater than 180°) were studied. As a rule, spacecraft designers prefer the type I trajectories since they are characterized by shorter trip times and smaller communication distances than the type II trajectories. In certain circumstances, however, type II transfers may prove to be desirable. Hence, both were examined.

Energy requirements vary markedly from one launch opportunity to another. Therefore, a rather wide range of payload capabilities results. Considering only type I trajectories over a 30-day interval in each opportunity, payloads for the Mars flyby mission were 11 400 pounds in 1971, 10 060 pounds in 1973, and 8920 pounds in 1975. The maximum communication distances required were 1.3 astronomical units in 1971, 1.34 astronomical units in 1973, and 1.69 astronomical units in 1975. Maximum flight times were 220 days in 1971, 196 days in 1973, and 209 days in 1975. Similar data are presented for type I and type II trajectories for all three mission profiles to Mars and Venus.

Author

INTRODUCTION

In order to achieve the scientific objectives of the space program, it is necessary to evaluate a variety of scientific missions over a wide range of mission variables. Such evaluations require a definition of vehicle payload capability and other significant mission parameters. The present study provides information of interest in the planetary program by examining Mars and Venus missions in the 1971 to 1975 time period.

References 1 to 6 present various analyses of one-way and round-trip missions to Mars and Venus. The three-dimensional, patched conic technique, described in references 1, 5, and 6, is also followed in this study. References 5 and 6 analyze one-way flyby missions to Mars for the 1966-1967 and 1969 launch opportunities, respectively, and payloads deliverable by the Atlas-Centaur launch vehicle are presented along with the trajectory parameters.

The present study is an effort to extend the knowledge of the trajectory requirements for the flyby mission through the Mars and Venus launch opportunities in the early 1970's and to apply the same type of analysis to the more complex orbiter and orbiter-lander missions for the same time period. Payload calculations presented herein are based on the capability of the Saturn IB-Centaur launch vehicle. A small retrostage was provided to perform propulsive maneuvers in the vicinity of the target planet when required. Mission trip time, communication distance at arrival, and energy requirements in the vicinity of Earth and the target planet are among the parameters that will affect the planning of missions. These parameters are shown as functions of launch date in this report.

The words payload and spacecraft are used synonymously to refer to the gross payload, which includes scientific instrumentation, spacecraft structure, and support equipment such as power and communications but does not include the retropropulsion stage.

The Mars opportunities in 1971, 1973, and 1975 are of primary interest; however, Venus opportunities in 1972 and 1973 are also included.

ANALYSIS

Three general mission profiles are considered in this study. The first, a flyby mission, requires that the spacecraft be injected near Earth onto a flight path that will intercept the target planet in its orbit. Spacecraft altitude at planet encounter is a function of the scientific objectives of the mission. If a lander rather than a flyby is desired, the flight path must be such that the spacecraft enters the planet's atmosphere and is slowed by drag until it is captured and impacts the planet.

The orbiter mission, the second mission profile considered, requires that the spacecraft, on nearing the target planet, be inserted into a given orbit about the planet. Data may then be obtained over a period of time from orbit altitude.

The orbiter-lander mission is a combination of the first two missions. On approach to the target planet, the lander portion of the spacecraft is separated from the orbiter, enters the atmosphere, and impacts the surface. The remaining portion of the injected weight contains a propulsive stage that inserts the orbiter spacecraft into orbit about the planet.

A patched conic, digital computer program, in which the motions of the planets are represented by noncoplanar ellipses, was used to generate the interplanetary trajectories required for this study. For a given launch date, the time of flight was varied to minimize the desired energy parameters for a given mission.

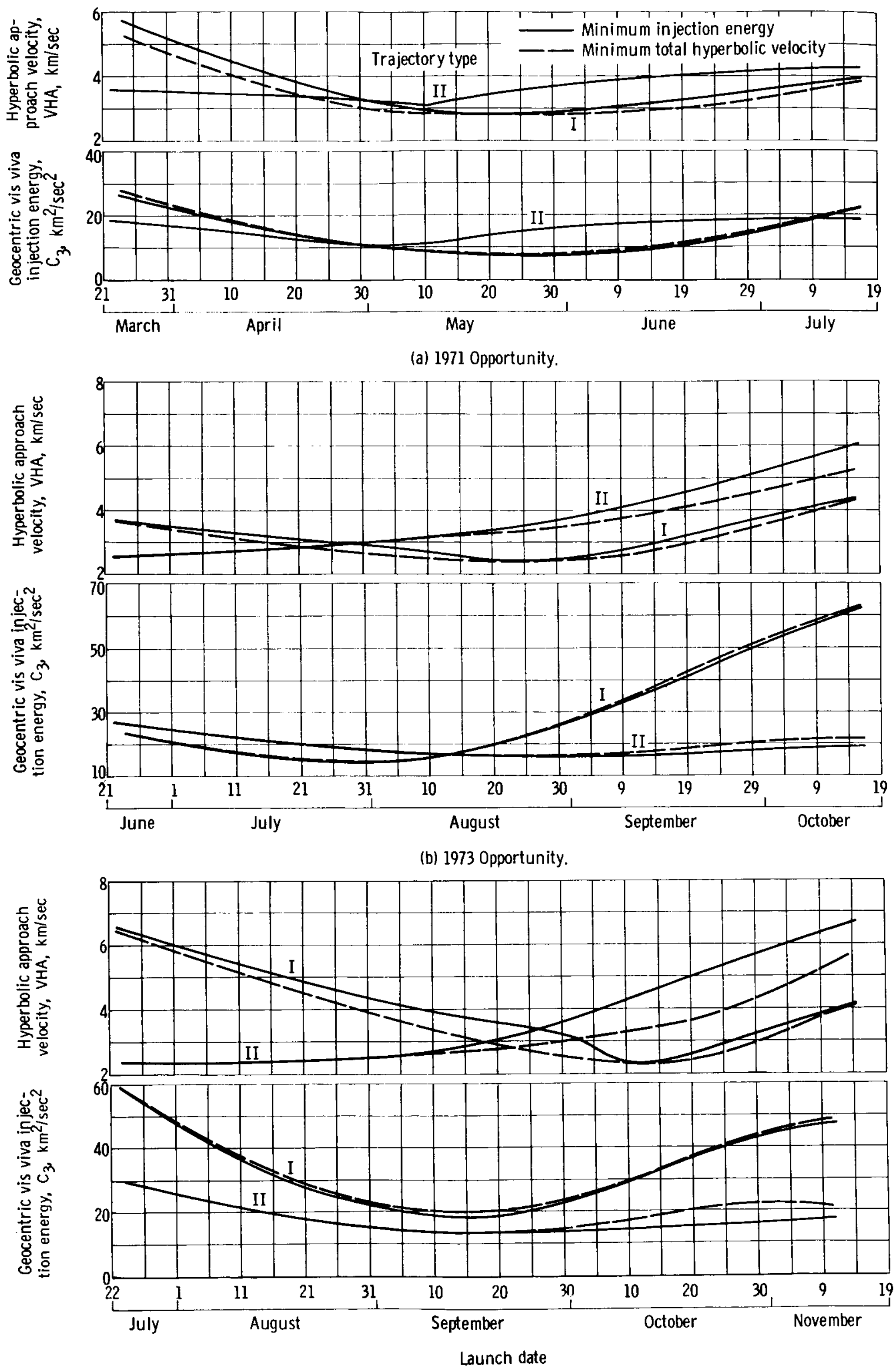
Since only the Earth injection portion of the flyby flights requires propulsion, the parameter minimized in this case was the injection vis viva energy C_3. Midcourse corrections for the flyby were assumed to be performed by the spacecraft itself.

The orbiter and orbiter-lander missions require, in addition to the injection phase of the flight, that a major velocity increment be removed from the spacecraft for insertion into orbit at the target planet. The insertion velocity increment is a function of the planet being considered, the hyperbolic excess velocity at arrival (VHA), and the orbit to be attained. Thus, for a given planet and a given orbit about that planet, the velocity increment is a function of VHA.

For convenience, the trajectories for the orbiter and orbiter-lander missions were based on minimizing the sum of the hyperbolic excess velocity at departure from Earth (VHD) and the hyperbolic excess velocity at arrival at the target planet (VHA). It is recognized that this simplifying assumption of using minimum total velocity does not yield the precise maximum payload. However, as is shown in the section RESULTS AND DISCUSSION, the payloads delivered are very nearly maximum. Although VHD was used as the Earth injection parameter in selecting the trajectories for the orbiter and orbiter-lander missions, C_3 ($VHD = \sqrt{C_3}$) is presented in all the figures as the injection parameter, to allow direct comparison with the flyby missions.

A 1000-nautical-mile circular orbit was selected for the Mars missions in this study. The insertion stage and its propellant capacity were sized for this Mars mission. In order to use the same insertion stage for Venus missions, which are characterized by higher insertion energy requirements, it was necessary to go to an elliptical orbit about Venus with a 1000-nautical-mile perifocus altitude and a 25 000-nautical-mile apofocus altitude.

The injection vis viva energy C_3 and the hyperbolic excess velocity at arrival are shown as functions of launch date for 1971, 1973, and 1975 Mars missions in figure 1 for both type I trajectories (heliocentric travel angle less than 180°) and type II trajectories (heliocentric travel angle greater than 180°). Figure 2 presents similar data for Venus missions in 1972 and 1973. The solid curves represent minimum vis viva injection

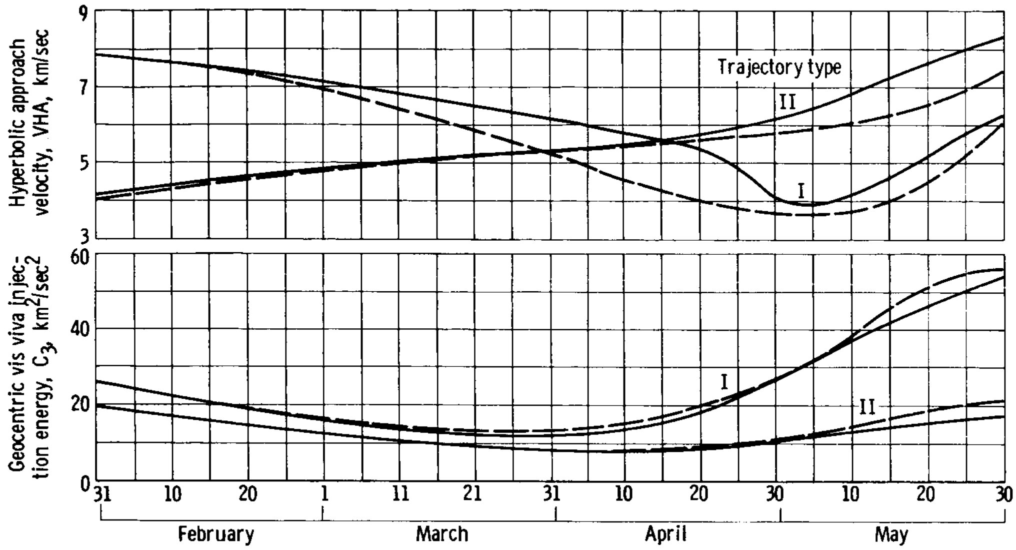


(a) 1971 Opportunity.

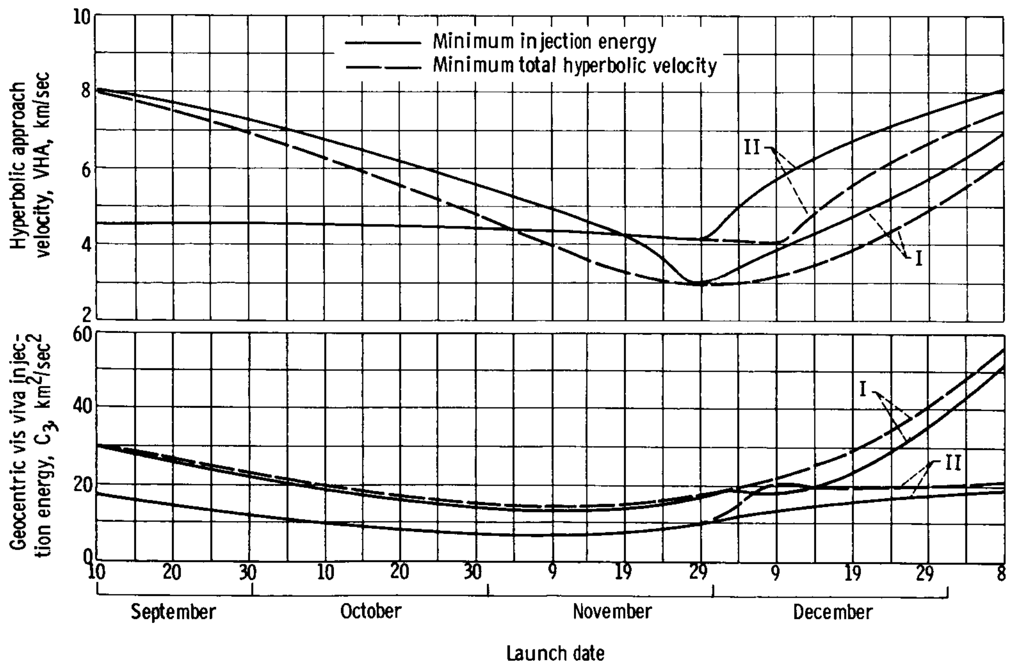
(b) 1973 Opportunity.

(c) 1975 Opportunity.

Figure 1. - Energy requirements for Mars trajectories.

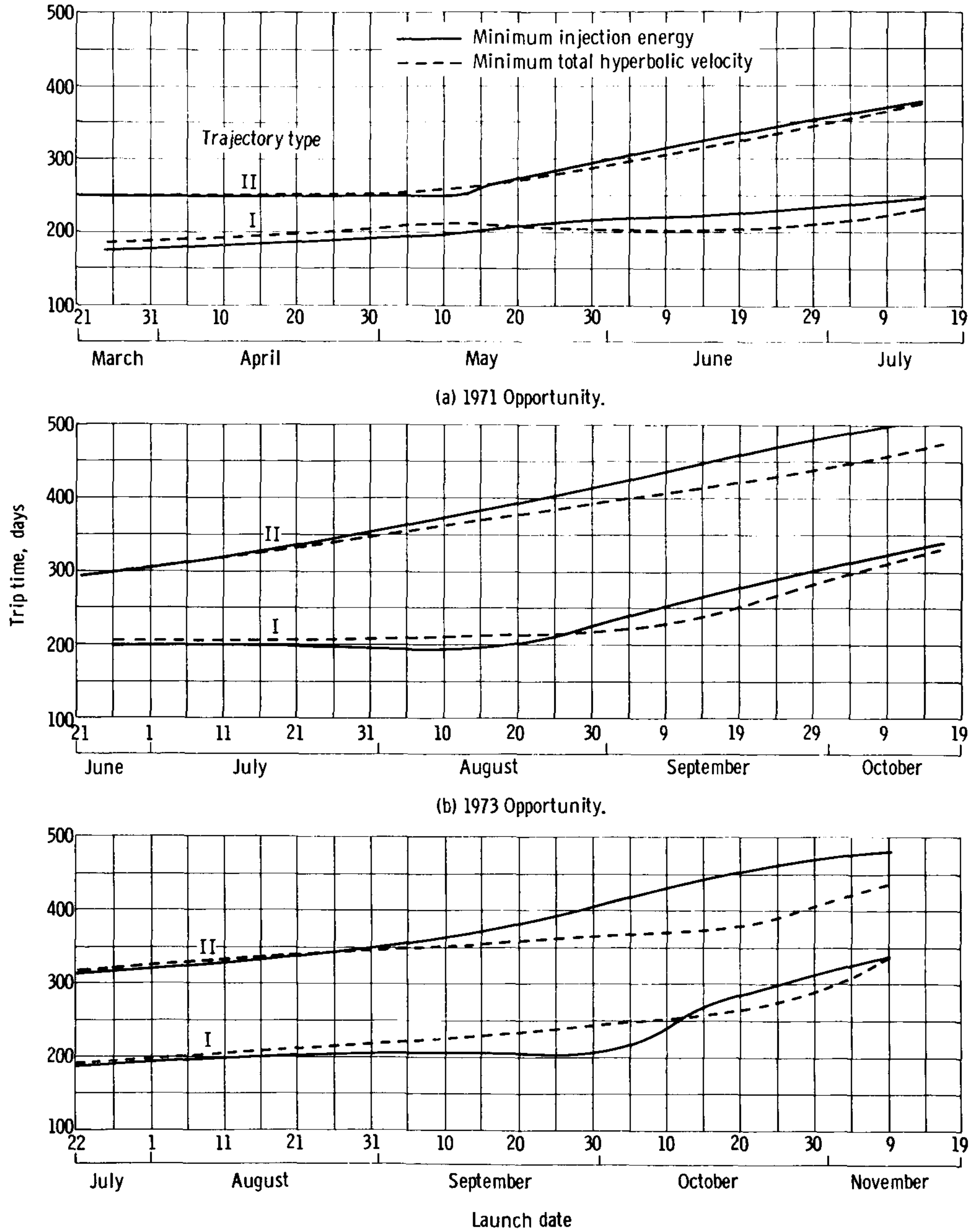


(a) 1972 Opportunity.



(b) 1973 Opportunity.

Figure 2. - Energy requirements for Venus trajectories.



(c) 1975 Opportunity.

Figure 3. - Trip time as function of launch date for Mars trajectories.

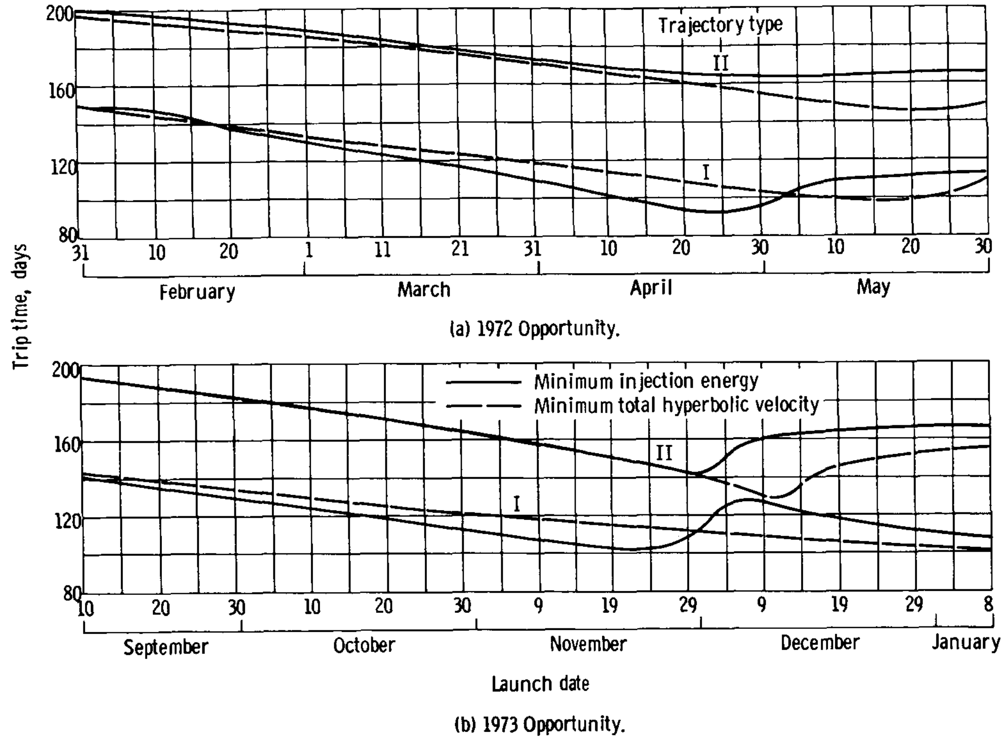


Figure 4. - Trip time as function of launch date for Venus trajectories.

energy trajectories, while the dashed curves represent trajectories for which the sum (VHD + VHA) was minimized. The minimum-injection-energy type II trajectories and the minimum-total-hyperbolic-velocity type II trajectories for 1971 coincide and are plotted as a solid curve in figure 1(a).

Trip time is plotted as a function of launch date in figure 3 for the Mars opportunities and in figure 4 for the Venus opportunities. Again, type I and type II trajectories are shown, and the solid curves represent minimum injection energy transfers, while the dashed curves are for minimum total hyperbolic velocity transfers.

For all payload calculations, the Saturn IB-Centaur launch vehicle was used to perform the launch-to-parking orbit and the parking orbit-to-injection portions of the flights. A 100-nautical-mile parking orbit was assumed.

An upper stage with the Earth-storable propellant combination monomethyl hydrazine and nitrogen tetroxide was used to perform the orbit insertion maneuver for orbiter and orbiter-lander missions. A specific impulse of 305 pound-seconds per pound was assumed for the propellant combination. The size of the insertion stage was based on a preliminary calculation of the maximum propellant required for the Mars missions. A 6000-pound propellant capacity was selected for the orbiter mission, and a 3000-pound capacity was selected for the orbiter-lander mission with a 5000-pound lander being separated before the insertion maneuver. The vehicle was assumed to carry only the

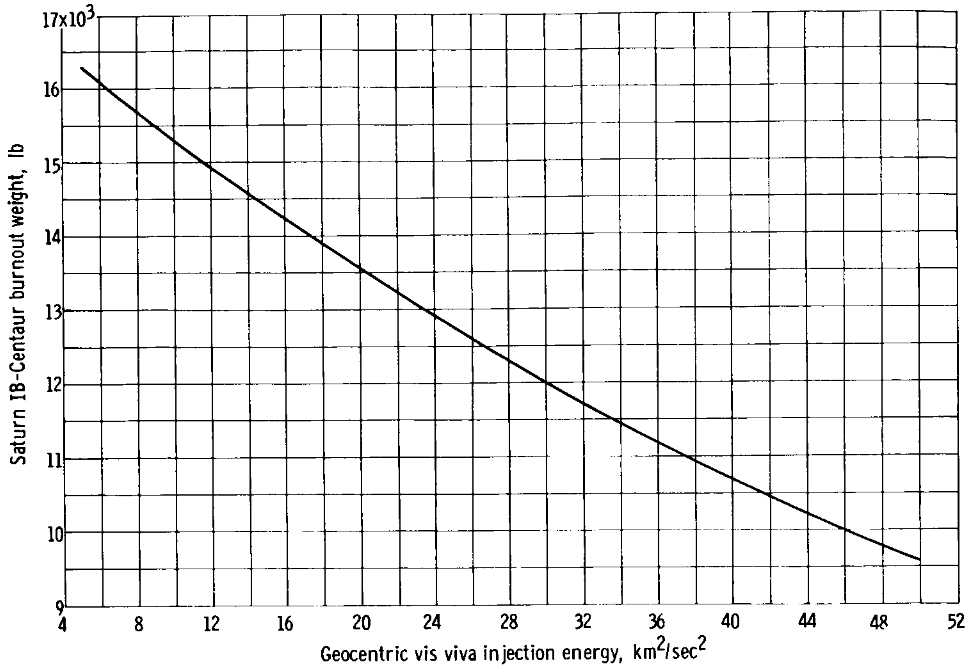


Figure 5. - Saturn IB-Centaur burnout weight as function of geocentric vis viva injection energy. Launch azimuth, 90°; Centaur jettison weight, 3987 pounds.

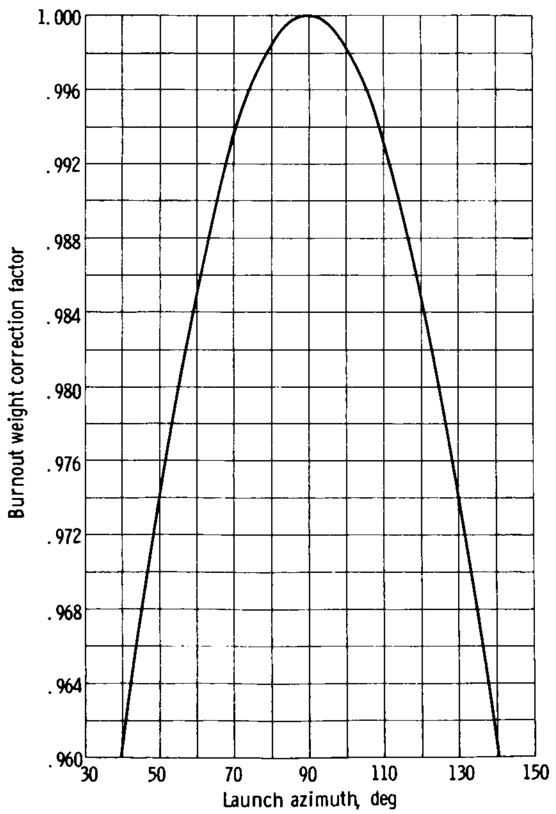


Figure 6. - Burnout weight correction factor as function of launch azimuth.

amount of propellants required to perform the mission. The hardware weight for the 3000-pound stage was based on reference 7, which presents a design for a similar stage to be used for probe-type missions in which only short space storage times are involved. Required modifications (e.g., addition of meteoroid shielding and increased attitude control propellants) were made to accommodate the long-term space storage associated with the interplanetary missions, and a final jettison weight of 1100 pounds was used. The 6000-pound stage was designed under the same ground rules as the smaller stage and had a jettison weight of 1250 pounds.

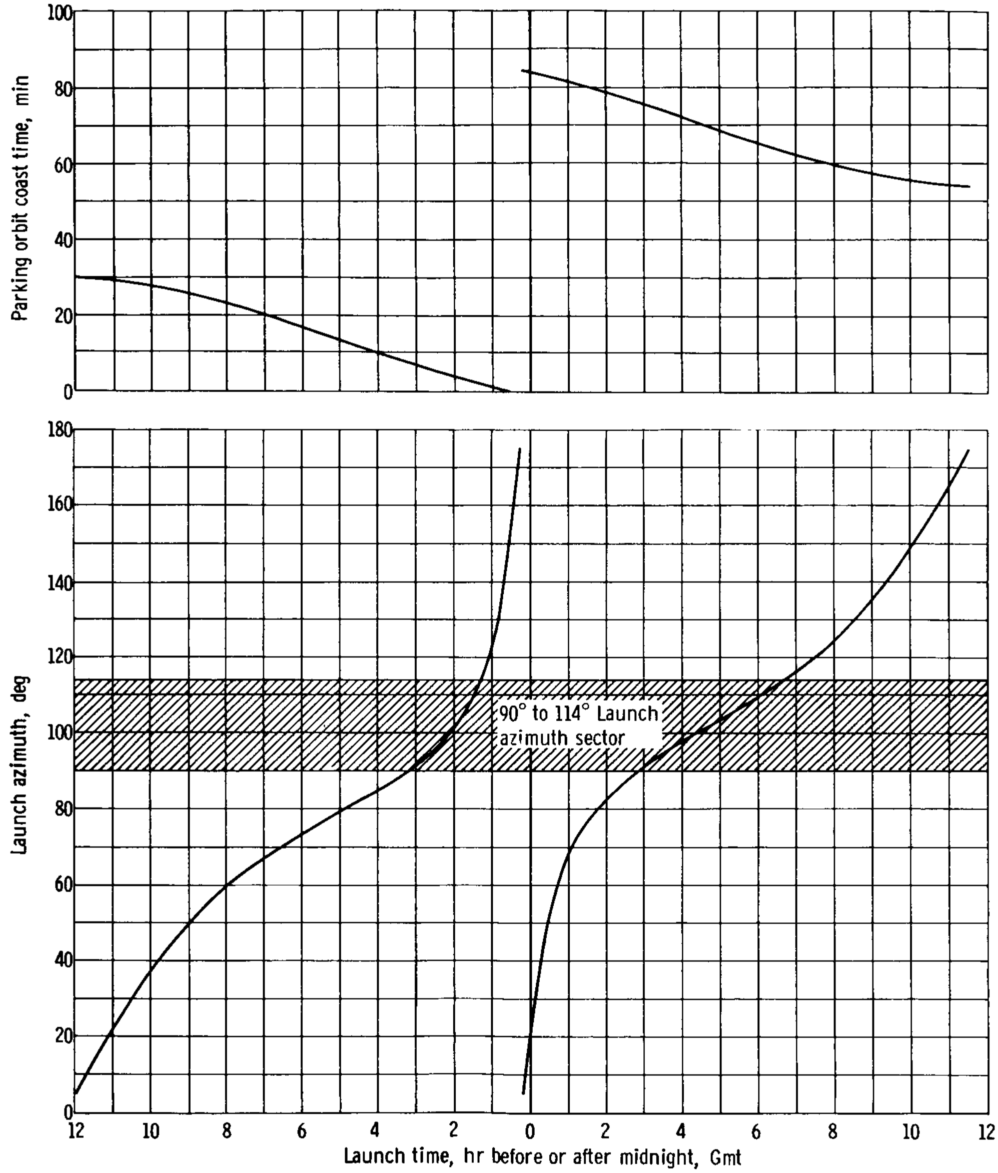
The cylindrical interstage between Centaur and the storable stage varied from 390 to 420 pounds according to the weight above Centaur. Adapters between Centaur and the spacecraft for flybys or landers and between the insertion stage and the spacecraft for orbiters and orbiter-landers were considered to be part of the payload since spacecraft configurations were not established.

The performance of the Saturn IB-Centaur launch vehicle was generated by integrating the flight trajectories on a digital computer. The booster flight profile was characterized by a short vertical rise followed by a zero angle of attack trajectory throughout the Saturn-IB-powered portion of the flight. An optimized steering program using calculus of variations was followed during the Saturn-IVB and the Centaur burning phases.

Launch vehicle burnout weight as a function of injection C_3 is shown in figure 5 for a due east launch (90° launch azimuth). Data from this curve may be corrected to other launch azimuths by multiplying the burnout weights shown by the burnout weight correction factor plotted in figure 6. Injected payload may then be found by subtracting the Centaur jettison weight, 3987 pounds. Weight data for Saturn-IB were based on information contained in reference 8. Centaur data were based on a proposed operational two-burn vehicle. A fixed flight performance reserve was assigned to Saturn-IB, and the nominal specific impulses of Centaur and the storable insertion stage were downgraded to cover dispersions in the assumed performance parameters.

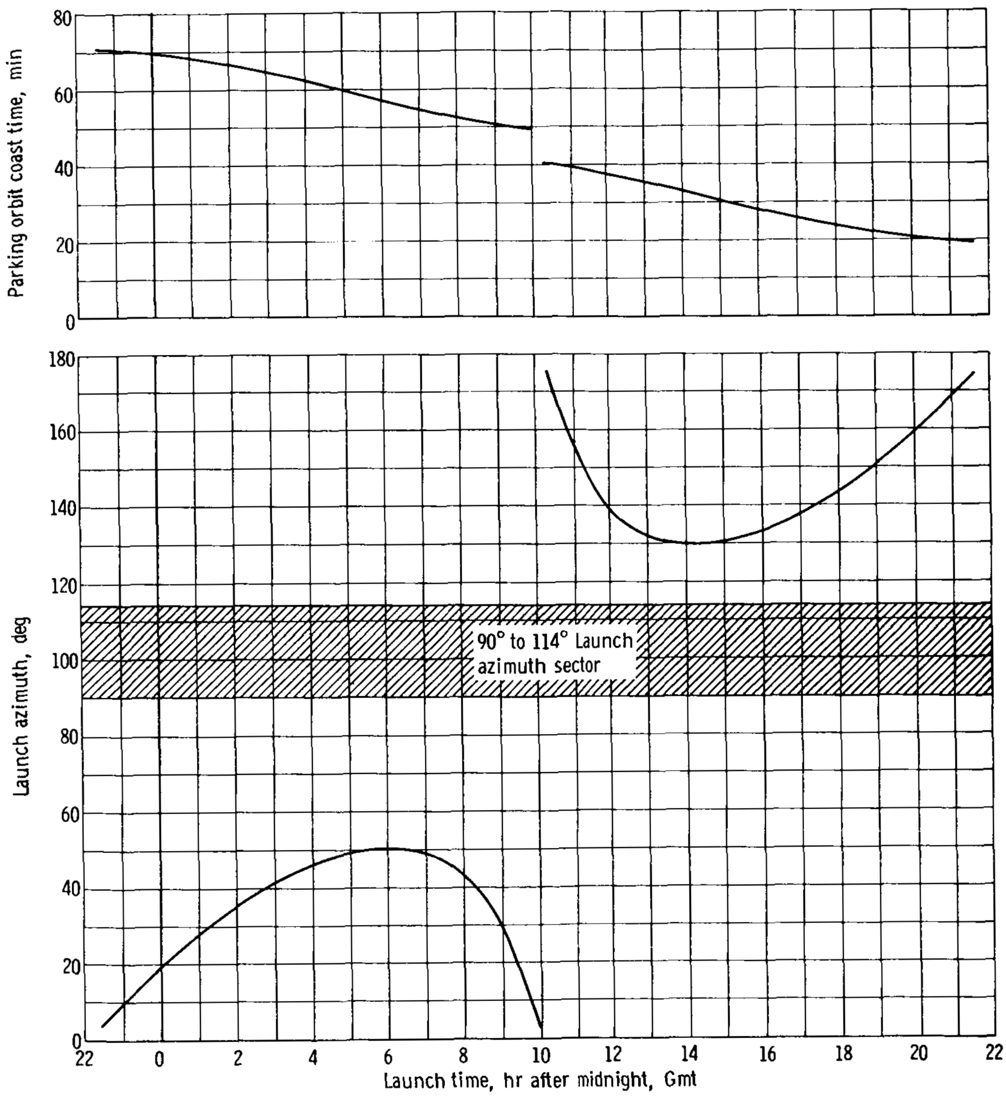
For a given launch azimuth, there will generally be two times during the day for which launch geometry requirements are satisfied, that is, two times when the phasing will be correct between the launch site and the aiming point in space (the outward radial).

Figure 7(a) presents the behavior of launch azimuth and parking orbit coast time as functions of launch time for minimum energy transfers on a representative day of a typical planetary launch opportunity. Observe that the launch azimuth curve has two branches. For an allowable launch between 90° and 114°, which was chosen to represent a typical range safety constraint, the launch window for one branch opens at 3 hours and 17 minutes before midnight and closes at 1 hour and 17 minutes before midnight. The launch window for the other branch opens at 2 hours and 54 minutes after midnight and closes at 6 hours and 40 minutes after midnight. Although either or both of these firing windows may be utilized, the one exhibiting the shorter parking orbit coast time is gen-



(a) May 23, 1971 trajectory.

Figure 7. - Parking orbit coast time and launch azimuth as functions of launch time for minimum-energy Earth-Mars trajectories.

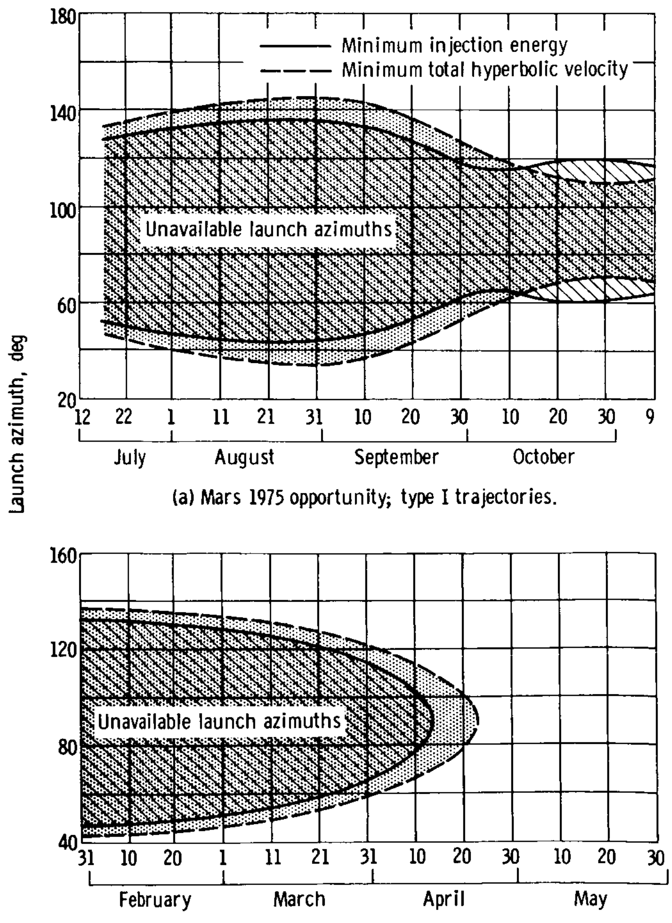


(b) September 15, 1975 trajectory.

Figure 7. - Concluded.

erally preferred from a launch vehicle standpoint.

On occasion, the launch geometry requirements cannot be satisfied for a particular launch azimuth sector. Such is the case for type I Mars trajectories in 1975 and type II Venus trajectories in 1972. This is illustrated in figure 7(b), which shows parking orbit coast time and launch azimuth as functions of launch time for minimum energy trajectories on a typical day in the 1975 Mars opportunity. Again, two branches of the launch azimuth curves are observed; the shapes, however, do not resemble those in figure 7(a). There is a launch azimuth sector (50° to 130°) for which there is no solution to the launch geometry problem for the minimum-energy trajectory. If the range safety launch azimuth constraint of 90° to 114° must be adhered to, launch windows for days such as this are nonexistent for minimum-energy transfers. For other than minimum-energy transfers, the launch azimuths required may be moved nearer to the desired range, but a severe payload penalty can occur, as is demonstrated in the section RESULTS AND DISCUSSION.



(a) Mars 1975 opportunity; type I trajectories.

(b) Venus 1972 opportunity; type II trajectories.

Figure 8. - Launch azimuth restrictions for 1975 Mars and 1972 Venus missions.

The launch azimuth restrictions for minimum-energy trajectories to Mars in 1975 and to Venus in 1972 are shown in figure 8. Minimum-injection-energy (flyby or lander) azimuth restrictions are crosshatched, and minimum-total-hyperbolic-velocity (orbiter and orbiter-lander) azimuth restrictions are shaded.

For those days during which a launch between 90° and 114° launch azimuth was possible, the performance of the launch vehicle was adjusted to a 114° launch azimuth by using the burnout weight correction factor shown in figure 6. In this way, a payload deliverable over the longest launch window possible within the range safety constraints was obtained.

For trajectories requiring launch azimuths outside the 90° to 114° range, the launch azimuth was selected on the basis of a 60-minute window. From figure 7(b), for example, a minimum launch azimuth of about 49.5° provides a 60-minute window

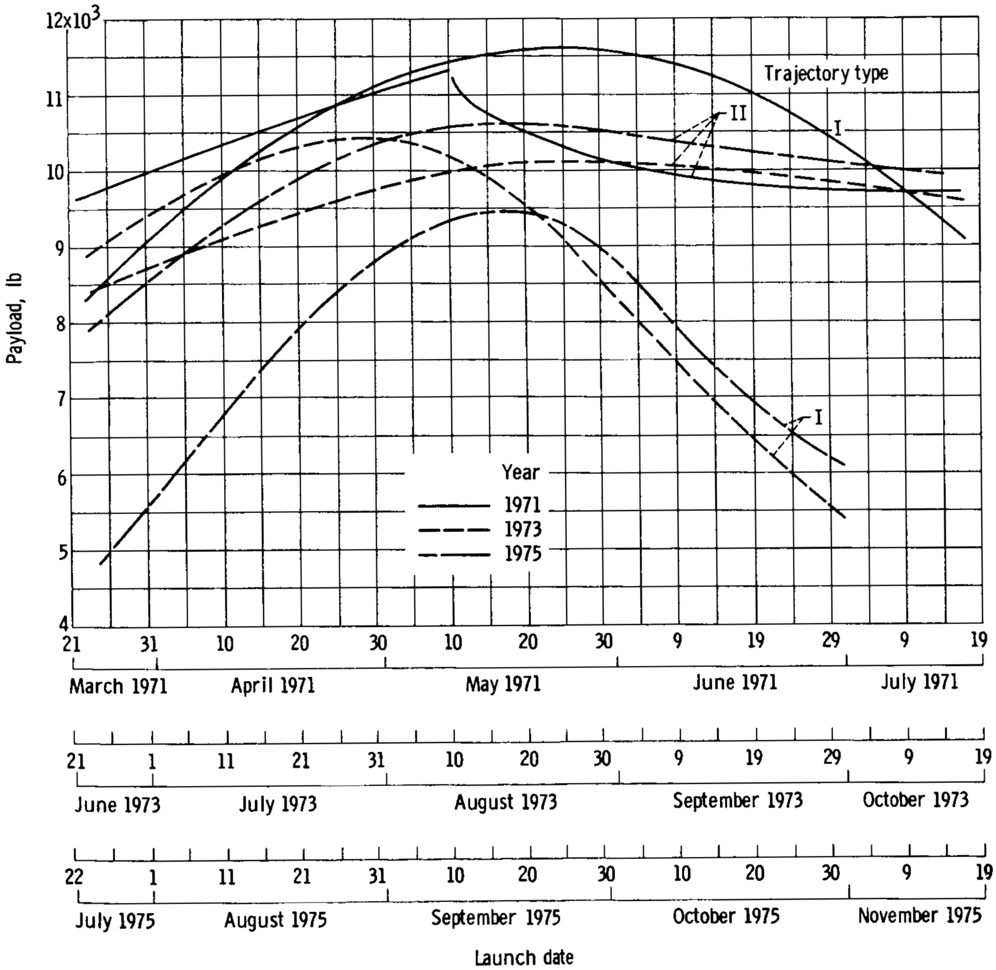


Figure 9. - Mars flyby payload capability as function of launch date.

for the day shown. Burnout weights from figure 5 were adjusted to the proper launch azimuths by the burnout weight correction factors for days requiring launch azimuths outside the assumed safety limits.

RESULTS AND DISCUSSION

Data were generated over approximately 120 days for each opportunity considered. A 30-day interval giving maximum payload was selected for each opportunity, and the results will be presented on the basis of these intervals. If longer or shorter launch intervals are desired, the required data may be obtained from the curves by inspection.

Payload capability, mission flight time, and communication distance at arrival are presented for the mission profiles investigated for Mars opportunities in 1971, 1973, and 1975 and for Venus opportunities in 1972 and 1973. Results for both type I and type II trajectories are presented.

Flyby Mission Profile

Figure 9 presents the Mars flyby payload capabilities for the 1971, 1973, and 1975

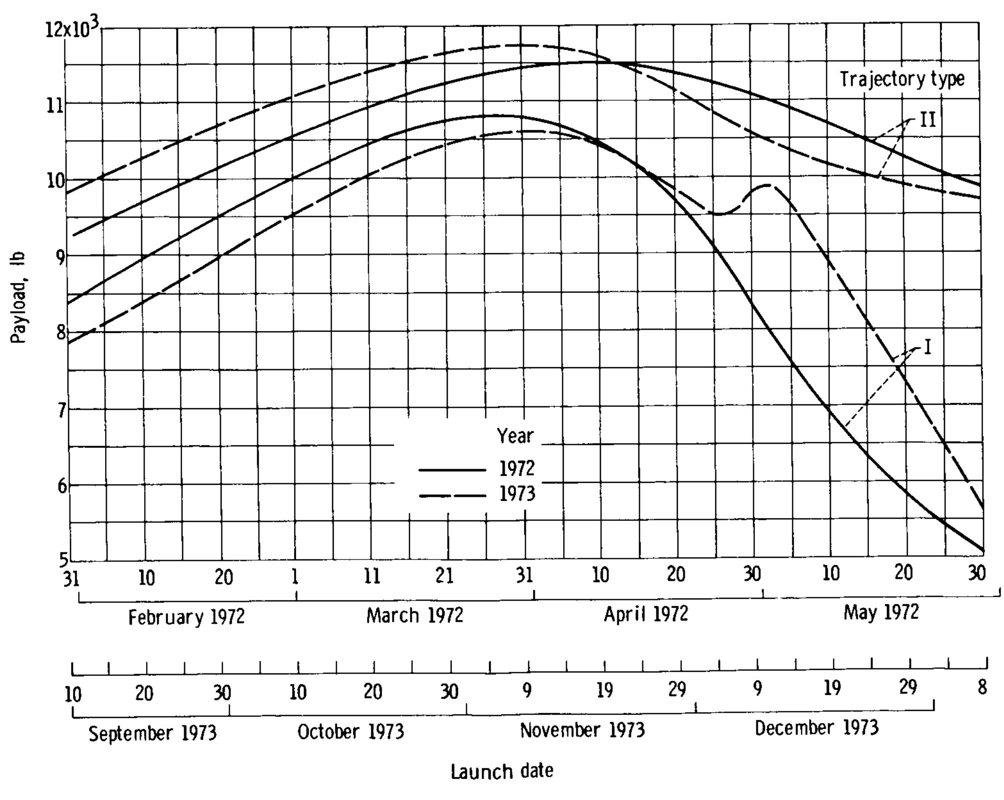


Figure 10. - Venus flyby payload capability as function of launch date.

launch opportunities. It should be noted that the payloads presented are gross payloads that have not been penalized for a spacecraft adapter or midcourse correction.

Based on a 30-day launch interval, Mars payloads near 11 400 pounds appear feasible in 1971 if type I trajectories are used. Use of type II trajectories in 1971 results in a payload capability of about 10 600 pounds.

In 1973, type I trajectories provide a 10 060-pound payload potential, and the type II trajectories yield about 10 030 pounds. It can be noted in figure 9 that, in 1973, a sizable launch interval could be realized by utilizing type I trips early in the opportunity and type II trips later in the opportunity with very little, if any, sacrifice in payload capability.

In 1975, the type I trajectories are characterized by high launch energies, and consequently, Mars payload capability is only 8920 pounds. Furthermore, launch azimuth requirements for the type I trips in 1975 are unusual, as evidenced from figure 8(a), and their use may be prohibited from a range safety standpoint. Payload capability for the type II trips in 1975 is nearly 10 460 pounds, and it may be desirable to accept the longer trip times and communication distances of the type II trajectories in order to achieve the higher payload capability and to avoid the launch azimuth difficulties associated with the type I trips.

Venus flyby payloads for the 1972 and 1973 opportunities are shown as functions of launch date in figure 10. The payload potentials for the type I and the type II trips to Venus in 1972 are 10 490 and 11 310 pounds, respectively.

In 1973, use of the type I trajectories yields a 10 230-pound payload potential as compared with 11 490 for the type II's.

It should be noted that for the type II trips in 1972, a launch azimuth problem similar

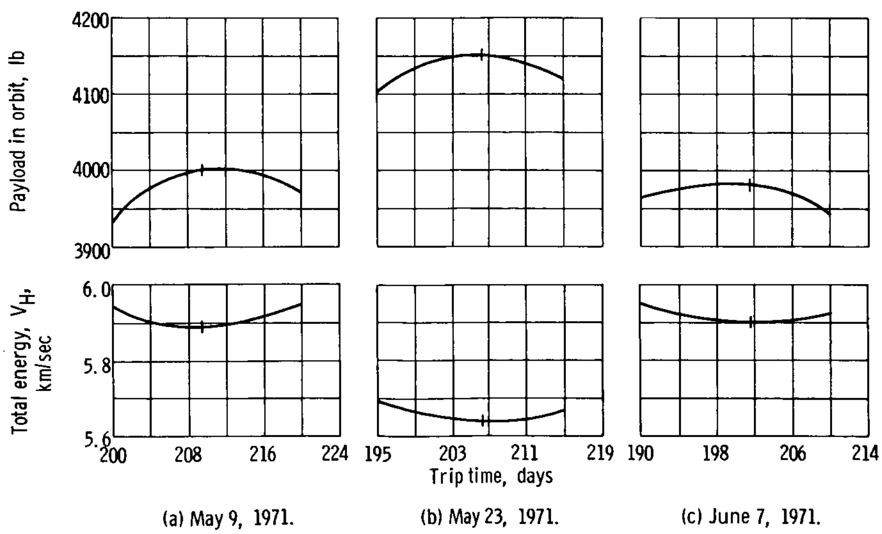


Figure 11. - Orbital payload and total energy requirement as functions of trip time for opening, middle, and closing days of 30-day Mars launch interval in 1971. 1000-nautical-mile circular orbit.

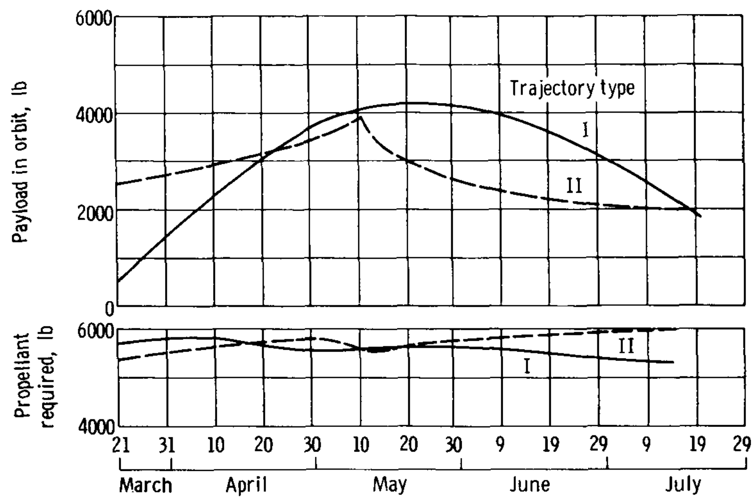
to that noted for 1975 Mars type I trajectories exists early in the launch interval, as shown in figure 8(b). This problem might be solved either by selecting type I trajectories in this portion of the interval or by shifting the type II launch interval toward the end of the opportunity. The latter solution would reduce the type II payload capability to about 11 000 pounds, which still exceeds that delivered by the type I trips.

Orbiter Mission Profile

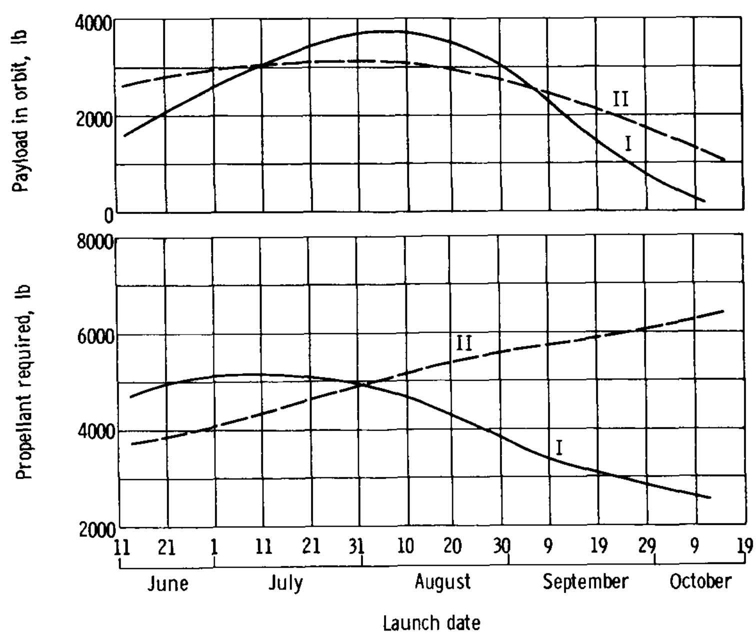
For the Mars orbiter mission, an orbit of 1000-nautical-mile altitude was selected. In order to determine whether or not minimizing the sum of the hyperbolic excess velocity at Earth departure and the hyperbolic excess velocity at planet arrival would maximize payload, figure 11 was constructed. Since trip time was varied to find the minimum total hyperbolic velocity (VH), the trip time was fixed at a few days greater than and a few days days less than the minimum total hyperbolic velocity trip time. Payloads were calculated for the fixed trip times and for the minimum total hyperbolic velocity trip time, and the results were plotted in figure 11. The total VH(VH = VHD + VHA) and the payload in orbit are shown as functions of trip time for the opening, middle, and closing days of a 30-day interval in the 1971 Mars opportunity. The ticks on the curves indicate minimum VH. While it may be seen that the maximum payload does not necessarily occur at minimum VH, the payload loss due to this assumption can be considered negligible for this study. This same type of analysis was applied to other opportunities with similar results.

Payload in orbit and the insertion stage propellant required for Mars orbiter missions in 1971, 1973, and 1975 are presented in figure 12. Note that all missions did not require a full propellant load; hence, the stage was not filled to capacity for these cases. Solid curves indicate type I trajectories, and dashed curves indicate type II trajectories. If type I trajectories are used in 1971 and a 30-day launch interval is assumed, a payload of about 3960 pounds can be orbited, while type II trajectories under the same assumption yield only about 3190 pounds of orbited payload. Propellant requirements are nearly constant throughout the opportunity for both types of trajectories. In 1973, type I and type II trajectories can deliver about 3470 and 3030 pounds, respectively, for a 30-day interval. The payload superiority of type I trajectories in 1973 decreases as the interval is extended until, for launch intervals greater than 55 days, type II trajectories deliver the higher payloads. For a 30-day interval in 1975, type II trajectories can deliver the heavier payload, placing 3670 pounds in orbit as compared with 2540 pounds for type I trajectories. If the launch interval is extended, the difference becomes even greater. For example, for a 60-day interval, type II trajectories deliver about 3200 pounds to orbit, while type I trajectories deliver only about 1580 pounds.

Because of the large velocity increments required for insertion into a 1000-nautical-

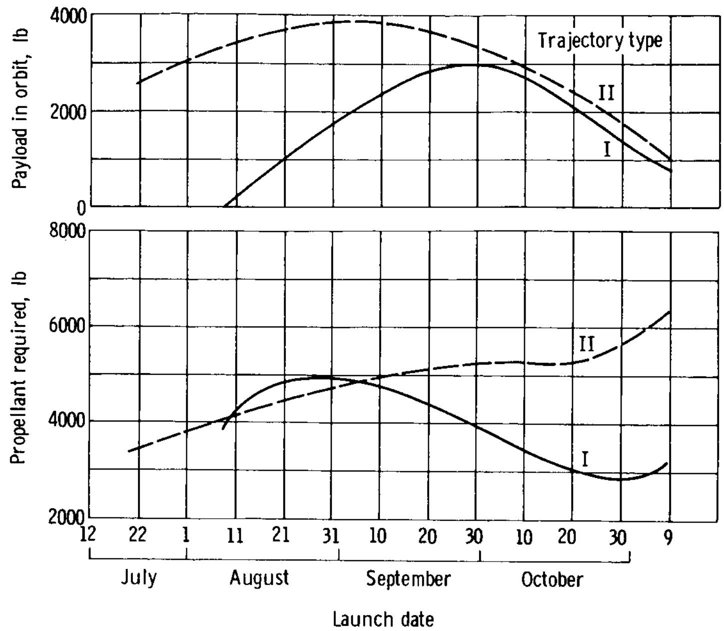


(a) 1971 Mars orbiter mission.



(b) 1973 Mars orbiter mission.

Figure 12. - Payload and propellant required as functions of launch date for Mars orbiter missions. Nominal kick stage propellant capacity, 6000 pounds; 1000-nautical-mile circular orbit.



(c) 1975 Mars orbiter mission.

Figure 12. - Concluded.

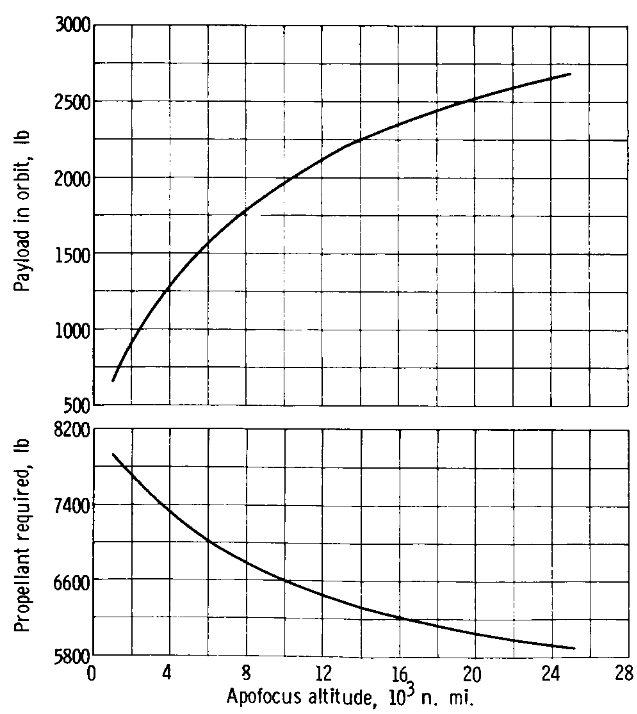


Figure 13. - Payload in Venus orbit and propellant required as functions of apofocus altitude for March 11, 1972 to show effect of elliptical orbits. Nominal kick stage propellant capacity, 6000 pounds; perifocus altitude, 1000 nautical miles.

mile orbit at Venus, payloads calculated for this orbit were very low, while insertion stage propellant requirements were high. In order to use the same insertion stage for Venus missions that was used for Mars missions, elliptical orbits were considered. Holding the perifocus altitude at 1000 nautical miles and raising the apofocus altitude reduced the velocity increment required so that, as indicated in figure 13, for a 1000- by 25 000-nautical-mile orbit the 6000-pound propellant capacity stage could be used and reasonable payloads could be obtained. Orbits lower than 1000 nautical miles were not considered in this study.

Figure 14 presents the payload in a 1000- by 25 000-nautical-mile orbit and the insertion propellant required as functions of launch date for Venus orbiters in 1972 and 1973. Payloads of about 3730 and 4000 pounds, respectively, can be delivered by type I and type II trajectories for a 30-day interval in 1972. For the same length interval in 1973, orbiter weights of 4690 pounds and 5100 pounds can be delivered by type I and type II trajectories.

Orbiter-Lander Mission Profile

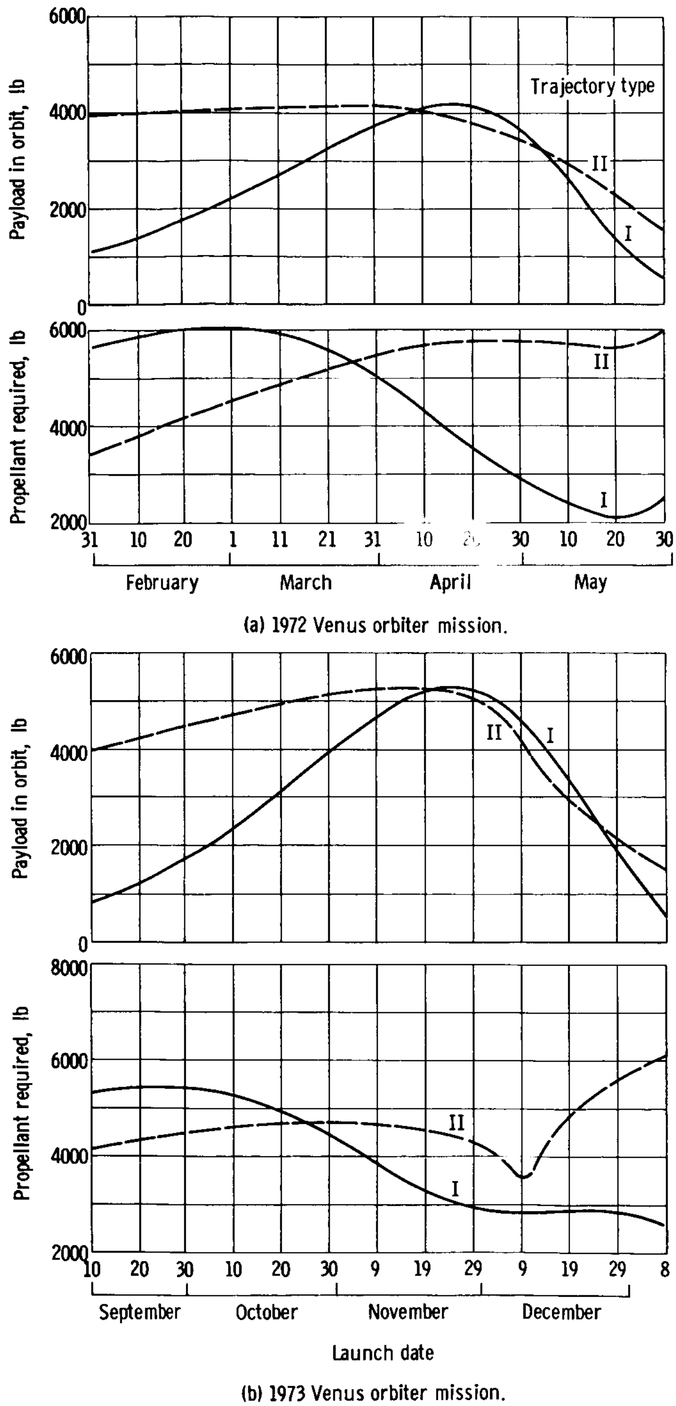
The same trajectory selection criterion that was used for the orbiter mission was applied to the orbiter-lander mission. In the case of the orbiter-lander, the lander, which for this study was assumed to weigh 5000 pounds, was separated prior to the insertion maneuver, thereby reducing the propellant required in the insertion stage. The nominal insertion stage propellant capacity for the orbiter-lander missions to both Mars and Venus was 3000 pounds.

Payload in orbit and insertion propellant required for Mars orbiter-lander missions in 1971, 1973, and 1975 are shown in figure 15 for both type I and type II trajectories. If a 30-day interval is sufficient for this mission and type I trajectories are selected in 1971 and 1973 and type II trajectories in 1975, payloads in orbit of 1110 to 1700 pounds are possible.

Similar data for Venus missions in 1972 and 1973 are shown in figure 16. For the same length interval, a Venus orbiter-lander mission can put 1680 and 2330 pounds in orbit in 1972 and 1973, respectively, with type II trajectories. However, type I transfers with their shorter flight times and smaller communication distances will deliver 1170 and 1680 pounds in 1972 and 1973, which are comparable to the Mars payloads.

Communication Distances

Figures 17 and 18 present communication distance at arrival as a function of launch



(a) 1972 Venus orbiter mission.

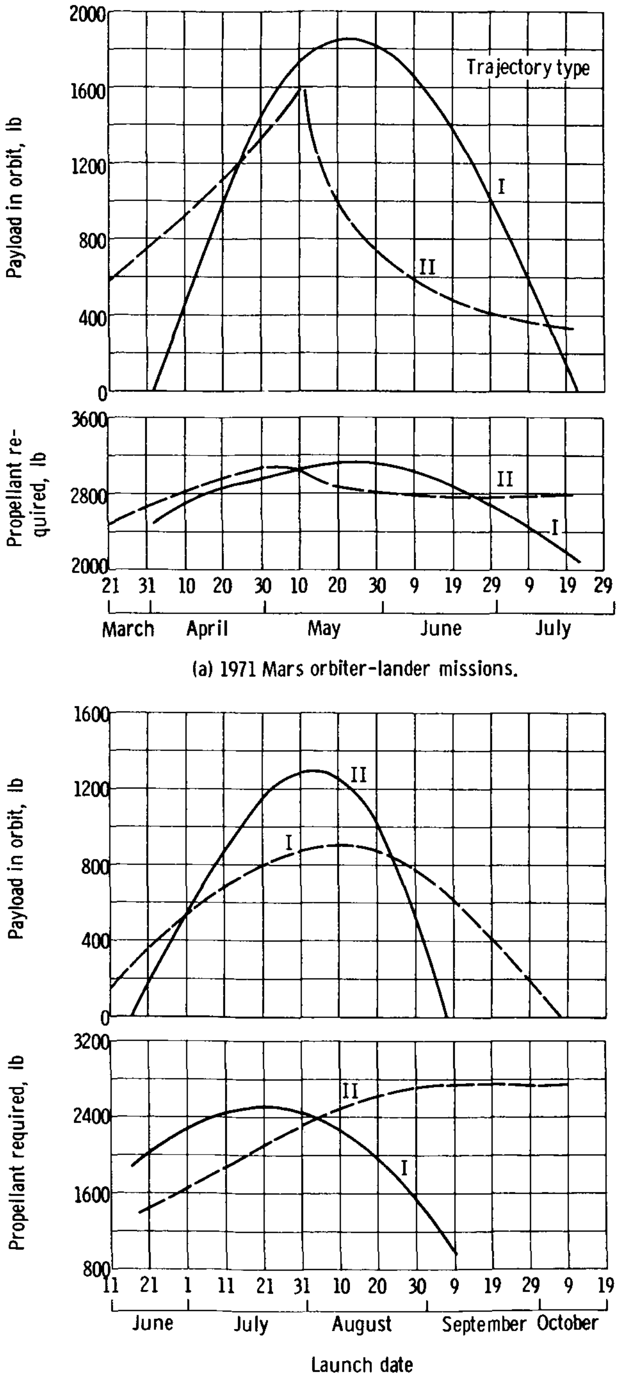
(b) 1973 Venus orbiter mission.

Figure 14. - Payload and propellant required as functions of launch date for Venus orbiter missions. Nominal kick stage propellant, 6000 pounds; 1000- by 25 000-nautical-mile elliptical orbit.

date for the Mars and Venus opportunities, respectively. Data for minimum-injection-energy trajectories and minimum-total-hyperbolic-velocity trajectories are shown for both type I and type II trajectories. In general, communication distances for type I trajectories are less than those for type II. However, near the end of the 1973 and 1975 Mars opportunities, the type I communication distances do become greater than the type II distances. Communication distances at arrival vary from about 1.14 to 2.61 astronomical units for Mars and from about 0.42 to 1.01 astronomical units for Venus for the 30-day launch intervals previously discussed in this section. Although the launch intervals herein were selected on the basis of payload, it is conceivable that communication distance criteria could, in practice, influence the selection of the interval.

Launch Azimuth Considerations

A 90° to 114° launch azimuth sector provides launch window capability for all trajectories considered, except the 1972 type II Venus and 1975 type I Mars trajectories. The latter exhibit peculiar geometry problems precluding the use of launch azimuths near 90°. Figure 8(a) presents the launch azimuth restrictions for the 1975 type I Mars trajectories, and figure 8(b) presents the lauch azimuth restrictions for the 1972 type II Venus trajectories.



(b) 1973 Mars orbiter-lander mission.

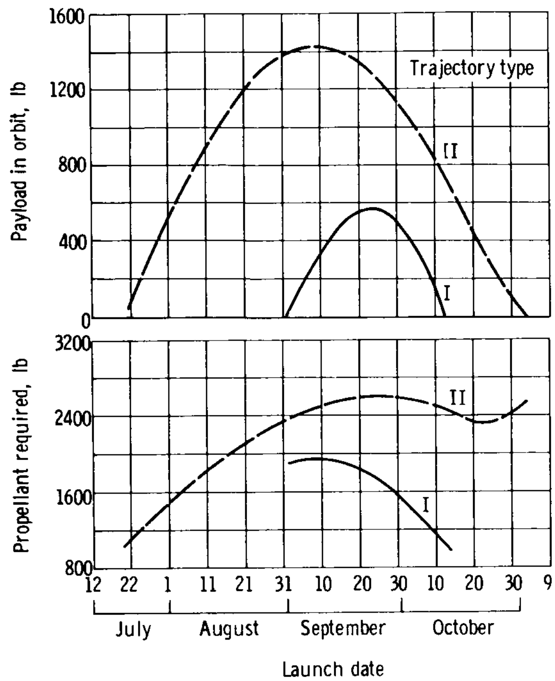
Figure 15. - Payload and propellant required as functions of launch date for Mars orbiter-lander missions. Lander weight, 5000 pounds; nominal kick stage propellant capacity, 3000 pounds, 1000-nautical-mile circular orbit.

Generally, spacecraft designers prefer type I trajectories, so that launch azimuth problems for the type II Venus trajectories in 1972 may be inconsequential. If type II trajectories were used for this opportunity, the launch interval could be shifted to April or May to alleviate the launch geometry problem (see fig. 8(b)). The payload capability for the new launch interval will be less, the exact amount depending on the interval selected and the mission considered. Figures 10, 14(a), and 16(a), which present payload as a function of launch date, illustrate these tradeoffs.

For the 1975 type I Mars trajectories, launch azimuth restrictions persist over the entire launch interval, as shown in figure 8(a). Efforts to shift the launch interval are ineffective, and payload capability diminishes quickly outside of the given launch interval, as indicated in figures 9, 12(c), and 15(c).

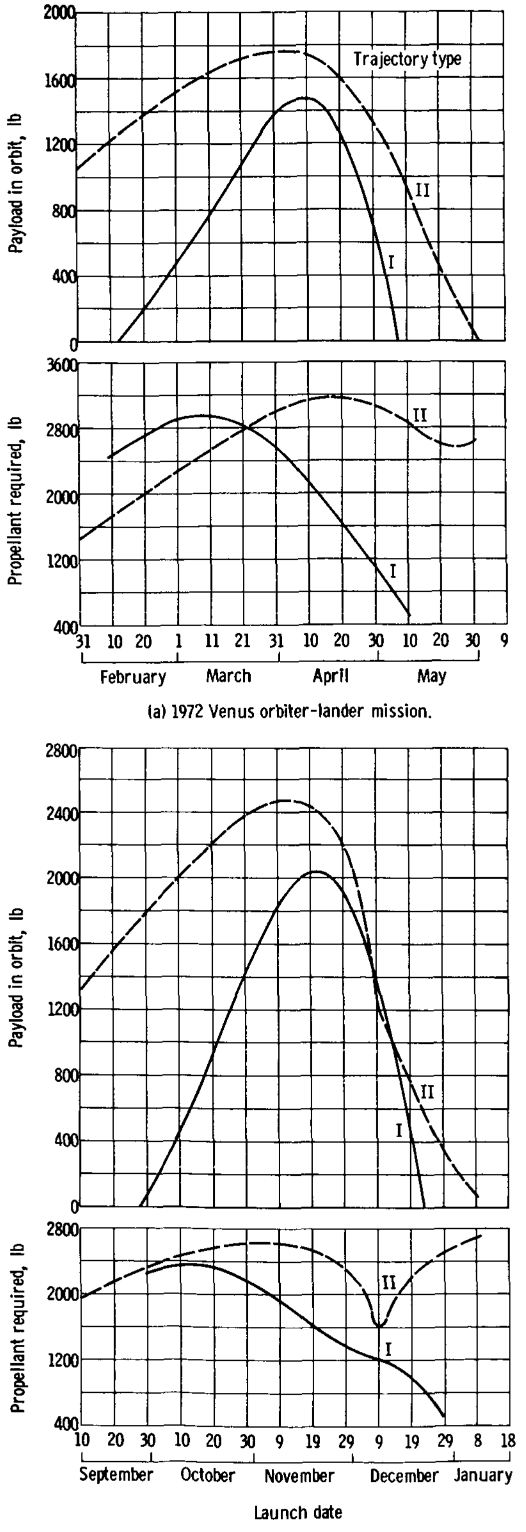
The trajectories thus far have been selected on the basis of minimum injection energy (or minimum total hyperbolic velocity). This results in a unique launch azimuth requirement. The launch azimuth requirement can, however, be varied by departing from minimum energy at the expense of payload. As an example, payload is shown as a function of launch azimuth for a 1975 Mars flyby mission in figure 19 and for an orbiter mission in figure 20. Typical opening, middle, and closing days for 30-day launch intervals are shown in the figures. Observe that, although departures from minimum energy allow some shift in azimuth, 90° launch azimuths are never fully achieved, and this type of tradeoff is quite costly.

For the 1975 Mars opportunity, either a range safety waiver for launch azimuth is required or type II trajectories should be utilized.



(c) 1975 Mars orbiter-lander mission.

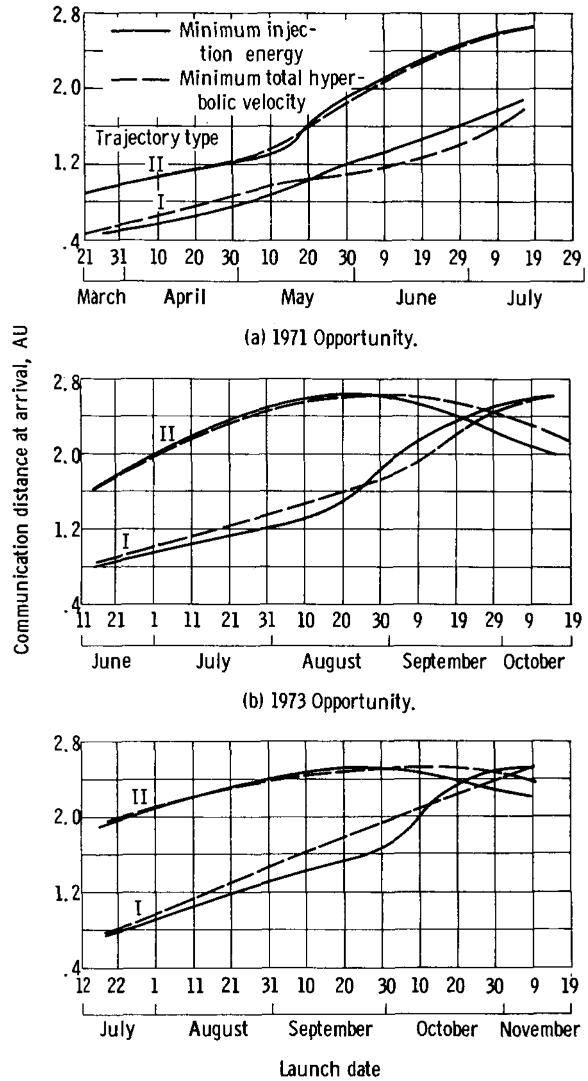
Figure 15. - Concluded.



(a) 1972 Venus orbiter-lander mission.

(b) 1973 Venus orbiter-lander mission.

Figure 16. - Payload and propellant required as functions of launch date for Venus orbiter-lander missions. Lander weight, 5000 pounds; nominal kick stage propellant capacity, 3000 pounds; 1000- by 25 000-nautical-mile elliptical orbit.



(a) 1971 Opportunity.

(b) 1973 Opportunity.

(c) 1975 Opportunity.

Figure 17. - Communication distance at arrival as function launch date for 1971, 1973, and 1975 Mars missions.

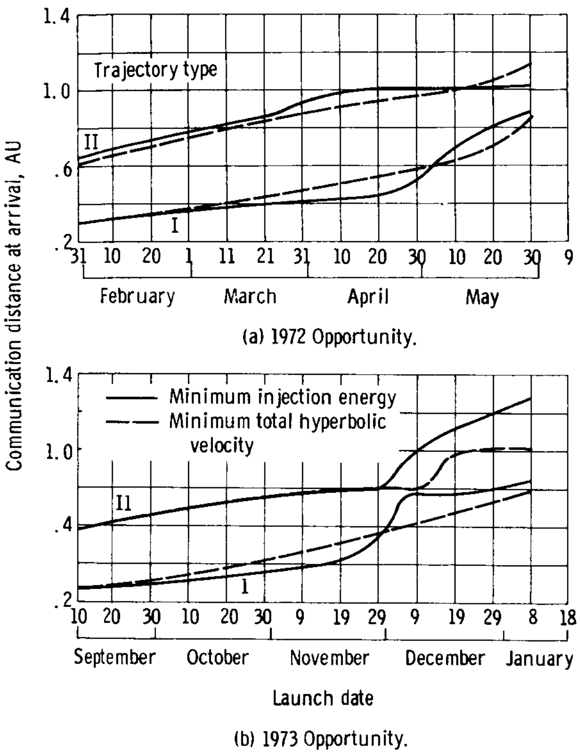


Figure 18. - Communication distance at arrival as function of launch date for 1972 and 1973 Venus opportunities.

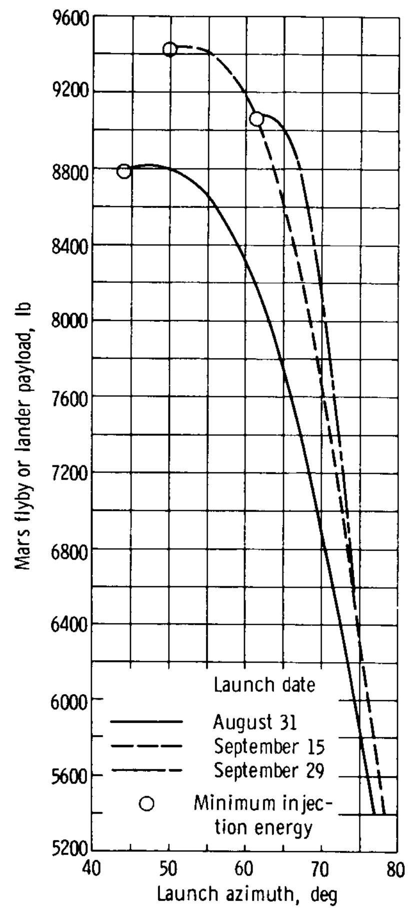


Figure 19. - Payload-launch azimuth tradeoff for 1975 type I Mars flyby trajectories.

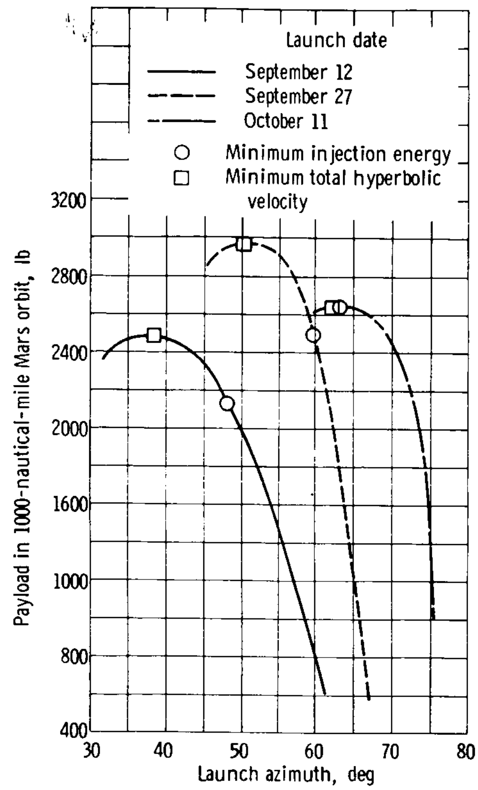


Figure 20. - Payload-launch azimuth trade-off for 1975 type I Mars orbiter trajectories.

TABLE I. - PLANETARY FLYBY PAYLOAD CAPABILITY OF SATURN 1B-CENTAUR

[30-Day launch interval; minimum injection energy.]

Year	Type of trajectory	Launch interval	Payload, lb	Maximum trip time, days	Maximum communication distance, AU
		Mars			
1971	I	May 10 to June 8	11 400	220	1.30
	II	Apr. 18 to May 17	10 600	266	1.46
1973	I	July 14 to Aug. 12	10 060	196	1.34
	II	Aug. 14 to Sept. 12	10 030	444	2.60
1975	I	Sept. 2 to Oct. 1	8 920	209	1.69
	II	Sept. 5 to Oct. 4	10 460	416	2.50
		Venus			
1972	I	Mar. 12 to Apr. 10	10 490	128	0.42
	II	Mar. 24 to Apr. 22	11 310	177	1.01
1973	I	Oct. 25 to Nov. 23	10 230	116	0.46
	II	Oct. 24 to Nov. 22	11 490	167	.79

TABLE II. - PLANETARY ORBITER PAYLOAD CAPABILITY OF SATURN 1B-CENTAUR WITH STORABLE INSERTION STAGE

[30-Day launch interval; minimum total hyperbolic velocity.]

Year	Type of trajectory	Launch interval	Payload, lb	Maximum flight time, days	Maximum communication distance, AU
Mars (1000-nautical-mile circular orbit)					
1971	I	May 9 to June 7	3960	210	1.14
	II	Apr. 19 to May 18	3100	265	1.51
1973	I	July 23 to Aug. 21	3470	214	1.62
	II	July 13 to Aug. 11	3030	364	2.55
1975	I	Sept. 13 to Oct. 12	2540	255	2.12
	II	Aug. 21 to Sept. 19	3670	357	2.47
Venus (1000- by 25 000-nautical-mile elliptical orbit)					
1972	I	Mar. 31 to Apr. 29	3730	118	0.58
	II	Mar. 7 to Apr. 5	4100	182	.89
1973	I	Nov. 9 to Dec. 8	4690	118	0.62
	II	Oct. 29 to Nov. 27	5100	165	.80

TABLE III. - PLANETARY ORBITER-LANDER PAYLOAD CAPABILITY OF SATURN 1B-CENTAUR WITH STORABLE INSERTION STAGE

[30-Day launch internal; minimum total hyperbolic velocity; 5000-lb lander.]

Year	Type of trajectory	Launch interval	Payload, lb	Maximum flight time, days	Maximum communication distance, AU
Mars (1000-nautical-mile circular orbit)					
1971	I	May 9 to June 7	1700	210	1.14
	II	Apr. 19 to May 18	1080	266	1.54
1973	I	July 19 to Aug. 17	1110	212	1.57
	II	July 26 to Aug. 24	830	383	2.61
1975	I	Sept. 8 to Oct. 7	270	249	2.04
	II	Aug. 25 to Sept. 23	1290	360	2.48
Venus (1000- by 25 000-nautical-mile elliptical orbit)					
1972	I	Mar. 24 to Apr. 22	1170	122	0.55
	II	Mar. 17 to Apr. 15	1680	178	.93
1973	I	Nov. 5 to Dec. 4	1680	120	0.60
	II	Oct. 27 to Nov. 25	2330	167	.80

CONCLUDING REMARKS

In this study, typical booster (Saturn IB-Centaur) performance curves were utilized to evaluate payload capabilities for Mars missions in 1971, 1973, and 1975 and for Venus missions in 1972 and 1973. Tables I, II, and III summarize the results of this study for flyby, orbiter, and orbiter-lander missions, respectively. In discussing results in this report, a 30-day interval has been generally assumed; however, the launch interval actually required for interplanetary missions such as these has not been firmly established. It will depend on factors such as the number of launches required, the number of available launch pads, and minimum turnaround time between launches. Data have been presented that allow various length intervals to be studied.

Type I trajectories appear to be favorable for Mars missions in 1971 and 1973 because of the high payload capabilities. The trip times and communication distances associated with these type I trajectories are also desirable. For 1975 Mars mission, the type II trajectories appear more desirable from both payload and range safety standpoints. Communication distances and flight times, however, are relatively high for these type II trajectories.

Type II trajectories deliver the higher payloads for both the 1972 and 1973 Venus opportunities. Weights comparable to those for Mars missions can be delivered to Venus, however, by using type I trajectories.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, February 25, 1966.

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