EXECUTIVE OFFICE OF THE PRESIDENT

NATIONAL AERONAUTICS and SPACE COUNCIL

WASHINGTON

October 18, 1963

Dear Mike:

Enclosed is a copy of the report on Future Unmanned Exploration of the Solar System, which I had indicated to you over the phone was almost complete. You can readily note the occasions where use was made of planetary gravity fields for trajectory shaping. I would appreciate your comments on those items specifically, as well as any comments you choose to make on the report as a whole.

I appreciated very much receiving your letter last summer. It is not very often these days that anyone takes the time to write a letter of that nature under those circumstances. It made me feel a little more that the battle is sometimes worth the effort.

We will surely have a chance to chat again sometime at a technical meeting or otherwise. In the meantime, I will, one of these weeks, get around to the question of the use of unconventional trajectories in manned operations, and will send you my opinions on the subject at that time.

Thanks once more for your letter of last summer.

Sincerely,

MWH

Maxwell W. Hunter, II
Member, Professional Staff

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Enclosure

FUTURE UNMANNED EXPLORATION OF THE SOLAR SYSTEM

By
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Washington, D. C.

Presented to
Executive Secretary
National Aeronautics and Space Council
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FUTURE UNMANNED EXPLORATION OF THE SOLAR SYSTEM

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INTRODUCTION

This paper presents a more comprehensive program for unmanned solar system exploration than is now planned. Certain of the data presented were obtained from recent studies, and some of the suggestions are already familiar to vehicle designers. Other information and viewpoints are my own, and need further verification or refinement by detailed study.

The basic reason for considering a more comprehensive unmanned lunar and planetary program is the fact that our current national effort is aimed almost completely at the Moon and the planets Mars and Venus. Actually there are 8 known planets other than Earth, 30 known natural satellites in addition to the Moon, and thousands of asteroids and comets in the solar system. It would seem, then, that we are expending a great deal of effort on only a small fraction of the solar system. This is the kind of situation which usually results in frequent program re-direction as the effects of advancing technology, with or without competition, inevitably raises questions of program adequacy. It could also lead to immature approaches if we should, for example, establish cooperative planetary probe programs with the Soviets without understanding the fundamental factors involved.

We have held a restricted view of our planetary program under the covering implicit assumption that the high performance rocket vehicles required for solar system-wide probing are so expensive that it just did not make sense to reach elsewhere in the system, at least for a long time to come. [This paper is directed toward proving this assumption to be false.] The solar system as an entity represents one of the few natural occurring space program package plans available. The distance to the nearest known star is 7,000 times the distance to Pluto, but Pluto is only about 26 times the distance of Mars. We shall examine this package.

VELOCITY REQUIREMENTS

To provide complete solar system coverage would require the ability to land instruments safely on the surface of every body, and to be able to go into every possible orbit about them. This means providing both enough spacecraft rocket braking to be able to land on any of the bodies without atmospheres, and atmospheric braking ability to land on those bodies with atmospheres. In addition, rocket impulse will be required for course changes and guidance corrections, and for braking into orbits. Furthermore, it would be extremely desirable to use atmospheric braking into orbit to reduce the burden on the rocket system. Each of these requirements must be analyzed in turn.

We shall start by examining the Earth launch velocity requirements for placing payloads at any location in the solar system. This will obviously require higher velocities than our current probes. Three separate aspects of launch velocities will be considered: (1) the ability to reach anywhere in the solar system including excess velocity to reduce travel times to distant targets, (2) excess velocity to open the launch windows to Mars and Venus so that much closer to year-round operations would be possible in those cases, and (3) out-of-ecliptic and solar probe missions. Normal trajectories will be considered first, but some effects of unconventional trajectories making use of the energy available from planetary gravitational fields will also be examined. In addition, payload velocity requirements to establish orbits about planets and to land on satellites will be explored.

The data presented in this paper assume circular, co-planar orbits for all planets and satellites. The satellite orbits were assumed to have a radius equal to the semi-major axis of the actual orbit. No attempt has been made to check the effects of these assumptions in the satellite cases, but accurately calculated data are presented in the planetary cases to indicate the validity of the approach. The pertinent characteristics of the planets and satellites in the solar system used in this memorandum are given in the Appendix. Significant uncertainties exist in some of these quantities, but they should not be great enough to change the basic conclusions.

LAUNCH VELOCITY

The basic launch velocity required is shown in Figure 1. The lower curve gives the launch velocity to reach the various planets with minimum energy expenditure. The travel times required are tabulated for each planet. In addition, curves of the velocities required to reduce flight time to the further planets are shown. As an example, minimum energy flight to the planet Uranus requires 52,000 fps launch velocity and takes 16.1 years, but this time could be reduced to 4 years by the use of 63,000 fps launch velocity. It is desirable to reduce the travel time to the far planets for several reasons. Reliability of equipment is perhaps the most obvious one. In addition, however, if one starts to think of flight times of 16 years (or 47 years to Pluto), one must consider a comparison of the development time plus vehicle travel time of the system under discussion with that of whatever new system will eventually replace it. This probably places a restriction on maximum flight time of the order of 10 years, as will be discussed later. I shall refrain from mentioning the importance of the political synodic period, which obviously has a major period of 4 years with a minor harmonic of 2.

Launch Windows

Excess velocity capability may be used to open the launch windows to Mars and Venus. Although Mars and Venus are closest to the Earth, they present the greatest launch window problem. The far planets are moving around the Sun so slowly that they act almost as fixed points, and in this case the synodic period approaches the Earth's revolution period of one year. For planets closer to the Sun than Venus, the synodic period becomes very short since the planetary orbit period becomes very small. A curve of synodic period with respect to Earth for all the planets in the solar system is shown in Figure 2.

The effects on launch velocity of 60-day launch windows to Mars and Venus are shown in Figure 3 using data from Reference 1. The data in this Reference were accurately machine-calculated including actual planetary orbital eccentricities and inclinations compared to the simplifications of Figure 1. The highest and lowest velocities shown for each pair of symbols represent respectively the worst and best synodic periods during the next 15 years. Only about 8 percent additional launch velocity is required in the worst actual case, a rather modest amount.

The definition of a launch window at even higher velocities is more complicated than might seem at first. Only the case of Mars will be discussed to illustrate the phenomena involved. The contours shown in Figure 4 are curves of travel time from Earth to Mars as a function of launch day for a total launch velocity of 60,000 fps.[2] The relationship between two succeeding synodic periods is shown utilizing the same contours as an approximation. Although it is possible to launch at any time of the year with 60,000 fps, there is a time (Point A) after which it makes more sense to wait Until Point B to launch, since the arrival time would be the same. Between Points A and B, we would be simply storing the probe in space, rather than on Earth. Thus, it can be seen that although a completely open arrival window is available, launching should occur only roughly half of the time.

Additional constraints are evident. A completely open arrival window requires a maximum flight time of 490 days and one might arbitrarily decide to limit this value to some smaller number. If so, both arrival and launch windows will be correspondingly reduced. A plot of both launch and arrival windows as a function of maximum flight time is shown in Figure 5.

At least one other limitation exists. If we make use of the completely open arrival window, then part of the time Mars will be on the opposite side of the Sun from the Earth. We will then either not be able to communicate with it and must store data for later transmission, or we must make use of a communication relay planetoid in solar orbit. We should establish such a communication relay planetoid at one of the Trojan libration points of the Earth-Sun system. Only one should permit continuous communication over the entire solar system at least as far in as Mercury.

In the meantime, it is desirable to know when during the synodic period this problem exists, and, accordingly, the band of time during which Mars is hidden from the Earth by the Sun is shown in Figures 4 and 5. We should, perhaps, limit the maximum flight time to about 280 days. This would avoid the problem of Mars being behind the Sun on arrival, and would mean arrival windows of approximately 53% of the synodic period and launch windows of 44%. These numbers are decreased approximately 10% if propulsive braking, rather than atmospheric braking is required at Mars.

A considerable investigation of high launch velocities for both Mars and Venus is required for a number of different synodic periods before these requirements can be accurately pinned down. It does appear, however, that 60,000 fps launch belocity will permit operation to Mars at least 40% of the time, which corresponds to roughly a one-year launch window. Venus should present an even more open launch window at comparable launch velocities. Thus, the use of launch velocities as high as 60,000 fps can vastly alleviate the launch window inconveniences of current programs.

Indirect Flight and Gravity Fields

The velocity requirement curves discussed to this point have assumed direct launch from Earth. It is possible, however, to make use of planetary gravity fields to deflect trajectories in such a way as to perform some missions with lower velocities. One example, of this is indicated in Figure 3 where data of Reference 3 are shown for Earth/Mercury and Earth/Mars missions utilizing a close flyby of Venus for orbit modification. Although the velocity requirements for Earth/Mars operations were not decreased significantly, it was possible to find Earth/Mars launch windows for every Earth/Venus launch window investigated. Since Venus launch windows are more frequent than those of Mars and rarely occur at the same time, this represents a more than doubling of the available launch windows to Mars.

In the case of Mercury, it is possible to reduce the velocity requirements by about 4,000 fps by means of the Venus flyby. Due to Mercury's high eccentricity and inclination, however, many of its launch opportunities require substantially higher values than shown. The values shown are typical of about 1/3 of the opportunities, or roughly one per year.

Other interesting unconventional trajectories exist. The major planets may be used to deflect trajectories to aid in close approaches to the Sun and out-of-ecliptic missions. Although a flight time penalty is involved in going farther away from the Sun to perform such missions, the velocity requirements are sometimes greatly reduced since the trajectories can be changed at aphelion with smaller velocity increments. Even the use of a second rocket impulse at aphelion without the benefit of a planetary gravity field will substantially reduce requirements.[4]

The use of Jupiter is particularly effective for several reasons. The planet is very large, and its gravity field is adequate for the necessary maneuvers. Jupiter's orbital velocity of 43,000 fps represents the magnitude of velocity to be deflected in most cases. For solar probe missions, this much velocity retrograde with respect to Jupiter will create a trajectory which hits the Sun, and the same amount deflected normal to the ecliptic will produce a trajectory which passes over the Sun at a distance equal to Jupiter's orbital radius. Examination of the trajectory mechanics shows that deflections of about 90^{o} are required, an obvious conclusion in the case of out-of-ecliptic trajectories. Since a close approach to Jupiter can deflect such velocities about 130^{o}, adequate margin is available. A further consequence of the large magnitude of Jupiter's gravity field is that the guidance accuracy required for the maneuvers is not great. For instance, an error of the order of 500 fps in hyperbolic excess velocity with respect to Jupiter produces only a one degree change in deflected angle. In addition to the beneficial effects of the large gravity field, Jupiter is close enough to the Sun that the flight time increases for the Jupiter flyby trajectories compared to direct flights are not excessive.

The use of a Jupiter flyby for solar probe missions is shown in Figure 6. The velocity required to come as close to the Sun as desired is only 50,000 fps if about 3 1/2 years' flight time can be tolerated as compared to a velocity of 80,000 fps required to approach to only 10 solar radii by conventional trajectories. The effect of the use of a second rocket impulse at the orbit of Jupiter is also shown in Figure 6.

An even more startling result occurs for out-of-ecliptic trajectories as shown in Figure 7. To launch directly from the Earth to 90° out-of-ecliptic and go over (or under) the Sun with a closest approach of one astronomical unit requires 140,000 fps. The same maneuver making use of a Jupiter flyby requires only 52,000 fps. Likewise, the requirement for going 90° out-of-ecliptic and making the closest possible approach to the Sun is reduced from 105,000 fps to 50,000 fps. The minimum energy curves of Figure 7 represent varying degrees of closest approach to the Sun. At low out-of-ecliptic angles, the probes stay essentially at planetary distance from the Sun since the magnitude of the orbital velocity of the planet is not changed appreciably at small angles. For out-of-ecliptic angles approaching 90°, the minimum energy probes pass very close to the Sun since the planetary orbital velocity must be nullified completely. In this case, the minimum probe velocity occurs at minimum additional velocity normal to the ecliptic.

The Jupiter gravity field can also be used to advantage for deep space missions beyond Jupiter, Figure 8. It is possible to escape completely from the solar system with an Earth launch velocity of only 47,000 fps as opposed to the 54,500 fps normally felt to be required. It is intriguing that the flight time to Pluto using this lower velocity requirement is only 25 1/2 years compared to 47 years without the aid of Jupiter gravity fields. When using Jupiter for trips to the other outer planets, the synodic periods of those planets and Jupiter create large intervals between launch windows which curtail the usefulness of this approach somewhat. The Jupiter/Saturn synodic period is almost 20 years while the other outer planet periods are about 13 years. For solar probe and out-of-ecliptic missions, however, the launch windows occur each Earth/Jupiter synodic period of just over one year.

It should be realized that the effects of using planetary gravity fields are large, not merely minor perturbations. Particularly in solar probe and out-of-ecliptic missions, velocity requirements are decreased by roughly factors of two and brought nicely into the range of other solar system requirements. This form of "gravity propulsion" is free energy, available in reliable form for the price of some clever guidance. It should be used as opposed to building needlessly high performance vehicles.

Asteroids and Comets

No attempt will be made to discuss the many different requirements created by the wide variety of orbits possessed by asteroids and comets. Asteroid velocity requirements will certainly fall within the velocities needed to cover all the planetary systems. Likewise, cometary velocity requirements will not be great if the orbit is known in advance to sufficient accuracy. In fact, by firing a probe out to the aphelion of a comet, it would be possible to match trajectories with only small velocity input as discussed with solar probe and out-of-ecliptic trajectories, and thus fly formation with the comet during its complete orbit of the Sun. Newly discovered comets are quite a different matter, as discussed in Reference 5. In that case, velocity requirements to intercept the comet after detection but prior to solar passage could become extremely large. If we rule out such comets of opportunity, the probe velocities required for the other solar missions discussed will be adequate for all asteroids and comets.

PAYLOAD VELOCITY

Figure 1 assumes either flyby missions, or the utilization of atmospheric braking into orbit or onto the surface of the target planet. It gives a feel for the Earth launch velocity required, but no indication of the braking problems experienced by the payload upon arrival. These braking requirements increase rapidly beyond 60,000 fps launch velocity. The large reductions in flight times to the further planets are achieved, obviously, because the probe is moving much faster in the deep space area.

An indication of the magnitude of this speed is shown in Figures 9 and 10 where the braking velocities required are shown for the launch velocities of Figures 1 and 3. It can be seen that these curves increase very rapidly at Earth launch velocities beyond the minimum. This is due to the fact that high speed rockets do not lose as much VELOCITY ~~energy~~ to gravity fields as low speed rockets, since they travel through a field more rapidly and hence are not decelerated for as long a time. This effect is compounded in solar space flight by the rapid traversing of both the Earth's gravity field and the solar field. This also explains why in Figure 10 a much larger spread occurs between the approximate curves and the actual calculations than in the launch velocities of Figure 3. It is interesting that, as expected, the Earth/Venus/Mars braking velocities are higher than Earth/Mars while the Earth/Venus/Mercury values are lower than Earth/Mercury.

Braking Within Gravity Fields

When braking is applied within a gravity field, an advantage is gained by the reverse of the process just described. In this case, the probe is accelerated by the gravity field until closest approach to the planet. If some velocity is removed at this point, the rate of travel out through the gravity field is reduced, and the field has time to extract more velocity than it put in during approach. The large planets, Jupiter, Saturn, Uranus, and Neptune have large gravity fields to aid in the efficiency of braking, and also extensive atmospheres which may be utilized. Pluto is farthest away, has a small gravity field, and no known atmosphere. Pluto thus likely represents the toughest target for landing missions of the next generation of probes.

Elliptical Orbits At Destination

It is frequently assumed when calculating the velocity requirements for establishing an orbit around another planet that the orbit will be circular at 1.1 planetary radius. In the case of a large planet, this results in high payload velocity requirements, and it is not clear that this is logical for two reasons. First of all, the surveillance of a planet may be done equally well and perhaps even better by a highly elliptical orbit with peri-apsis (point of closest approach to planet-perigee at Earth) sufficiently close to the planet. The velocity requirements for such orbits are far smaller than for the close circular orbit. Secondly, for a large planet like Jupiter, we may be even more interested in landing on its satellites. The closest large satellite of Jupiter is Io, and it is at 6 planetary radii. The establishment of a circular orbit at 1.1 planetary radius as part of the process of landing on a satellite at 6 planetary radii would be a waste of energy since the orbit must later be raised.

Atmospheric Braking

Atmospheric braking is extremely important as a means of decreasing total payload velocity requirements. Although atmospheric landing on the surfaces of the major planets may well be feasible, it will not be considered further here since a detailed analysis would be necessary. The magnitude of braking required for landing on the minor planets and for using the major planetary atmospheres as an aid in approach control will be considered.

Both Mars and Venus have extensive atmospheres, do not tend to have large braking requirements, and have been well investigated. The other two minor planets, however, have tricky braking problems. The relatively large braking requirements of Mercury and Pluto are shown in Figures 9 and 10. The Mercury requirement is large because of the high orbital speeds close to the Sun and its high inclination. The Pluto requirement is large if high launch velocities are used to decrease travel times. By coincidence, both braking requirements tend to be about 50,000 fps. The atmosphere of Mercury is estimated to be as dense at the surface as the atmospheres of Earth or Mars at about 150,000 feet. Since Mercury and Mars are small, the atmosphere of Mercury should be about as effective as the atmosphere of Mars except for touchdown requirements. It is known that the Martian atmosphere is much more effective than Earth's above those altitudes. In typical Martian entries, the velocity has decreased to far less than 10,000 fps by 150,000 feet altitude. Pluto is thought to be slightly larger than Earth, and might have an atmosphere which has remained undetected due to the great distance of observation. On the other hand, the very low temperature of Pluto may have frozen out the normal atmospheric gases, and the atmosphere of such a cold planet may be vastly different from Earth's. Thus, atmospheric capture at Pluto would be possible only if it has an atmosphere which is as useful for braking as that of Earth.

It is reasonable to send the first probes into orbit around the major planets using only propulsive braking if we do not feel confident enough about the use of atmospheric braking. The first probe could then survey the atmosphere in question and return enough information so that later probes could use atmospheric braking and land on the target satellite with propulsive braking. Personally, I think we might well use atmospheric braking at an early date. I believe in using what the Good Lord put there when it comes to flight mechanics, and both large gravity fields and atmospheres are fair game.

Approach To Satellite Orbit

There are many different ways of approaching a planet and landing on its satellites. The method described here is based on making the maximum use of planetary atmospheres and gravity fields to reduce the propulsive velocity requirements to a minimum. Upon first approaching the planet, the probe is assumed to make a close approach to the planetary surface and extract enough velocity by atmospheric braking to enter a capture orbit of very high eccentricity. The apo-apsis (point farthest from planet-apogee at Earth) of this orbit will be limited to about one hundred planetary radii which will result in an orbital period of less than one month for the major planets. The capture requirements for the major planets are shown in Figure 11. The very strong attenuation of the braking requirements of Figures 9 and 10 by the planetary gravity fields is evident.

If the launch velocity is limited to 60,000 fps then maximum atmospheric braking of 20,000 fps will cover all requirements. A velocity decrease of only 20,000 fps in an atmosphere as large as those of the major planets, even if we are moving at high speeds and aren't exactly sure of their composition, sounds much easier than killing 37,000 fps at a small planet like Earth on the way home from the Moon. The 20,000 fps represents also the maximum propulsive velocity required to establish the elliptical planetary survey orbits mentioned previously. For maximum Earth launch velocity of 54,500 fps, this requirement is reduced to less than 10,000 fps for all major planets, Figure 11.

After the probe is established in its capture orbit its manner of transfer to any final circular orbit would depend on the radius of final orbit. If the final orbit is less than about five planetary radii, the probe would atmospherically brake an amount of velocity upon next reaching peri-apsis such that the subsequent apo-apsis would be at the desired orbital radius. The velocity required to establish orbit must then be added when subsequent apo-apsis is reached. This amounts to using a Hohmann transfer from close planet approach to orbit, and is the method commonly employed to establish Earth orbits. If the final orbit is greater than five planetary radii, the probe would add an amount of velocity at initial apo-apsis to raise the peri-apsis to the desired radius, and then subtract the necessary velocity to establish orbit upon reaching peri-apsis. The payload velocities to be added or subtracted at final orbit injection for the major planets are shown in the curves of Figure 12.

Landing on the Satellites

The solar system bodies without appreciable atmospheres, as seen through the eyes of a propulsive system designer, are shown in Figure 13. This is a plot of escape velocity versus surface gravity. An idea of the size of these bodies may be obtained from the diameters shown in Figure 14. Ganymede and Titan are actually larger than Mercury, although Mercury has the largest escape velocity of 13,700 fps. An atmosphere has been detected on Titan, but it is probably not great enough to be helpful.

A group of satellites similar on the basis of Figure 13 to ours exists at 6,900 to 10,400 fps escape velocity. They are Io, Europa, Ganymede and Callisto, the four large satellites of Jupiter (called the Galilean satellites since they were discovered by Galileo); Titan, the large satellite of Saturn; and Triton, the large satellite of Neptune. Of the two dozen other known satellites in the solar system, Rhea and Dione of Saturn have the largest escape speeds, but they are only about 25% of those of the satellites above. The largest known asteroids are somewhat smaller than Rhea and Dione.

If we provide a spacecraft with 54% more surface g and 33% more escape velocity capability than required for our Moon, it will be able to land on all the bodies without atmospheres in the solar system including our Moon. This is an example of the relatively modest improvements required to obtain complete versatility in total solar system operations.

The spacecraft velocity requirements to match orbital speeds and land on the various satellites are also shown in Figure 12 under the assumption that the orbit matching maneuver takes place in close proximity to the satellite involved. It can be seen that although 16,000 fps would permit landing on almost all satellites of all planets, the Galilean satellites of Jupiter could not be reached. 18,000 fps would give us the ability to reach Callisto. Actually, it is possible that even Io, Europa, Ganymede and Jupiter V could be reached within 18,000 fps spacecraft velocity by means of unconventional trajectories utilizing a close flyby of Callisto just as a Venus flyby can reduce

Mercury braking requirements. No attempt is made here to check this case.

LAUNCH VEHICLE

The previous sections outline a wide variety of mission requirements, and it is possible to synthesize a number of different launch vehicle and spacecarft combinations to meet a significant fraction of these missions. Only one such combination will be presented here together with some of the reasoning behind its choice. Further study will be required along these lines. The vehicle presented, however, represents a reasonably well thought out starting point.

Vehicle Velocity Capability

If deep space travel times of the order of five years are acceptable, then the critical velocity design condition in the solar system apparently is a landing on Io, the inner Galilean satellite of Jupiter. This would require a spacecarft propulsive velocity of 26,000 fps and Earth launch velocity of perhaps 48,000 fps. The 48,000 fps was selected to somewhat reduce the travel time to Jupiter, but without expending very much velocity in this critical case since the travel time to Jupiter is only 2.69 years at minimum velocity. Requiring 26,000 fps in the spacecraft leads to a complicated design which would require a two stage braking rocket. Although this is quite reasonable, the suggested vehicle would be designed for only 18,000 fps in the spacecraft. This represents an extra 1,500 fps beyond the theoretical minimum for landing on Callisto shown in Figure 12, and should be an adequate allowance for guidance and control corrections.

The satellites of Jupiter which are within the orbit of Callisto (Ganymede, Europa, Io and V) can then be reached in one of two ways. The first is by utilizing a close flyby of Callisto as previously mentioned if this proves feasible. The second would be to supply extra propulsive braking at Jupiter by carrying the final stage of the launch vehicle to Jupiter to supply extra capability. This may be very tricky, and perhaps impractical, but it does represent an efficient vehicle utilization. In this case, an extra 8,000 fps must be provided so that the basic launch vehicle speed must be 56,000 fps. Thus the vehicle postulated has 56,000 fps earth launch capability with the assumption that spacecraft velocity will be used at launch if appropriate, and final stage launch vehicle impulse will be used at Jupiter for landing missions when and if feasible.

The critical launch missions are shown in Table I with an arbitrary six-year flight time limit on all missions and with the use of Jupiter flybys for solar probe, out-of-ecliptic, and deep space missions.

TABLE I

VELOCITY REQUIREMENTS

Payload Velocity = 18,000 fps

	Launch Velocity Required (fps)	Available from Payload (fps)	Launch Vehicle Velocity (fps)
Out-of-Ecliptic Via Jupiter	52,000	18,000	34,000
Solar Probe Via Jupiter	52,000	18,000	34,000
Jupiter/Io	48,000	-8,000	56,000
Jupiter/Callisto	48,000	0	48,000
Neptune/Triton*	57,000	6,000	51,000
Pluto Flyby*	64,000	18,000	46,000

*6 Years Via Jupiter, Approximately 9 Years Direct

Table II gives flight times for various deep space missions assuming a 56,000 fps launch vehicle capability.

TABLE II

DEEP SPACE FLIGHT TIMES

Launch Velocity = 56,000 fps
Payload Velocity = 18,000 fps

	Velocity Available From Payload (fps)	Total Launch Velocity (fps)	Flight Times (Years) Direct	Flight Times (Years) Via Jupiter
Jupiter/Io	-8,000	48,000	1.8	---
Jupiter/Callisto	0	56,000	1.0	---
Saturn/Titan	6,000	62,000	1.8	1.7
Uranus/Miranda	8,000	64,000	4.0	3.3
Neptune/Triton	6,000	62,000	7.0	5.1
Pluto Flyby	18,000	74,000	6.6	5.2

Vehicle Size

Since these appear to be high velocity requirements, it is natural to emphasize the use of the most modern of high energy chemical rockets. There is no particular reason why high energy rockets should cost any more per unit weight in production than current rockets once their development is complete and the new techniques understood. There is also no reason that they should not be used in first stages as well as upper stages. The stage velocities achievable with high energy hydrogen/fluorine rockets are shown in Figure 15. With a stage of average structural efficiency, we can generate about 27,000 fps with an initial weight/payload weight ratio of 10. Considerations of energy imparted to the payload, however, indicate that the optimum growth factor for any given stage is about 6.

We can use the curves of Figure 15 to get a rough feeling for the size of the vehicle required. Using a three-stage hydrogen/fluorine rocket, a ratio of launch weight to spacecraft weight of only 150 would be achievable with stage λ' of about 0.92. This λ' should be possible in all stages considering the density of hydrogen/fluorine, but requires modern structures for lightweight stage design. Thus, if a 3,000 pound spacecraft were carried, the launch weight of the vehicle would be 450,000 pounds, about the same as Titan II. A breakdown of stage weights and velocity increments is given in Table III. The spacecraft will be discussed later. If the third stage were usable for auxiliary spacecraft braking, it might be very difficult to achieve a λ' of 0.92 in this stage. Of course, if such a stage were also used on current vehicles, it might have to be designed for micro-meteorite and long term hydrogen storage requirements for Earth orbital missions. The effects of such requirements will not be considered here.

TABLE III

	Stage Weight (lb)	Total Weight (fps)	Velocity Increment (fps)	Initial Stage Wt. / Final Stage Wt.
Payload	500	500		
Spacecraft	2,500	3,000	18,000	6.0
Third Stage	12,000	15,000	20,000	5.0
Second Stage	60,000	75,000	20,000	5.0
First Stage	375,000	450,000	16,000*	6.0

*Assumes 6,000 fps drag, gravity, and nozzle losses.
Impulsive Velocity = 22,000 fps

The very interesting point evident is that proper staging and use of the high energy chemical propellants now available would permit the development of a probe vehicle for use throughout the solar system which is only one-third of the launch weight of Saturn 1-B. It is not necessary to use Saturn V or "Nova."

High Energy Vehicles Compared with Conventional

This report has purposely devoted much discussion to velocity requirements before considering vehicle design. This was done in an attempt to really understand the phenomenon involved because of a feeling that too much current thinking is based on computer calculations which bury a mixture of rocket performance with flight mechanics and make no attempt to understand the interaction of the two. For instance, the suggestion of a 60,000 fps probe is normally greeted as an expenditure of tremendous energy. We have used Atlas/Agena vehicles in escape missions of 37,000 fps, and they utilize conventional propellants. If all else is constant, the velocity of a rocket is directly proportional to its specific impulse, and the specific impulse ratio of high energy propellants to conventional is almost identical to the ratio of 60,000/37,000. Thus modern rockets, not monster rockets, are what is required and the extra energy needed is already contained in the high energy propellants.

As another example, it has been often stated that the launch window problems of Mars and Venus are an insurmountable fact of Nature. Yet the curves usually quoted to prove this are payload versus launch date curves for Atlas/Agena. Actually, that vehicle is very marginal for these missions, and has a very steep curve of payload versus velocity. It is the marginal vehicle more than Nature's barbs that shoot down the payload.

Another viewpoint is contained in References 1 and 3. The energies to perform various missions are plotted there, but these are either the energies required beyond orbital velocity, or beyond escape velocity. These are then inferred to be valid measures of difficulty and expense of mission. But the probe must be brought from the surface of Earth, and a measure of Earth launch energy is a more appropriate measure of expense. As an example, the increase in launch energy required beyond escape velocity for an increase from 40,000 fps to 47,000 fps launch velocity is almost a factor of 10. The difference in Earth launch energy, however, is less than 40%, not nearly as formidable a number.

The proposal here to use high velocities merely for convenience of probe operations, such as the reduction of travel time or opening of launch windows, is contrary to most thinking concerning the efficient utilization of rocket vehicles. It is normally felt that it is wasteful of money not to make use of the larger payload-carrying ability of the rocket at lower speeds. We use high speeds merely for convenience in all other transportation systems, however, as soon as we learn to produce such speeds reasonably.

Some interesting analogies exist between the use of high velocity rockets and the design of long range transport aircraft. A long range transport carries a great deal of fuel but only a small passenger or freight fuselage. It ignores the fact that at short ranges, it is theoretically capable of carrying very much greater loads. Of course, short range transports are also built, but only two or three range classes cover the total operation and each in its class is very versatile. Those that were carefully designed for just one mission disappeared a long time ago. The point to be made is that it has been found by experience to be economically infeasible to match carefully the fuselage size of the airplane to each mission. The long range missions must be flown, and the jobs to be done at short ranges do not justify the expense of the added complexity.

The suggestion is simply that the same approach be applied to space probes to cut down vehicle variations and permit the cost savings from the resulting standardization. Actually, the principal is even more applicable to rockets, since a three stage rocket is analogous to a squadron of three different airplanes. For solar probe and out of ecliptic missions, as an example, the entire spacecraft of the vehicle of Table III can be replaced by a 4,200 pound payload package. The two lower stages by themselves can place about 35,000 pounds in orbit. Most of the missions require the high velocities, but the same rocket can be used at low velocities without undue penalty. An aggressive try for a real transportation system, even without reuse, should yield much lower dollar per mission costs than the current systems.

A New Launch Vehicle

Investigations of possible vehicles to meet these requirements should include both all new vehicles and modifications of current vehicles such as adding modern staging to and upgrading Atlas and Titan II. I would prefer to see a new high-energy vehicle from the ground up with a very strong attempt to use modern techniques to pioneer low-cost launch operations. Such a vehicle should not require as many people to launch as a Scout. By making modest use of imagination, it should be possible to vacuum deposit the check-out instruments in the rocket with negligible weight penalty so that ground facilities are cut to an absolute minimum.

Although this may seem a radical suggestion, I am convinced that no serious attempt has yet been made to really apply modern electronic techniques to rocket checkout. Internal checkout is, of course, common airplane design practice. Furthermore, redundancy of design is very important in achieving high reliability right from the start, but the success of the technique depends on locating failures even when they do not cause flight failure so that components may be improved. This latter requires clever and discriminating telemetry, but not necessarily fantastic numbers of channels. With the capabilities inherent in microminiaturization, I think an integrated system of pad checkout and in-flight failure reporting could be achieved with great reliability and little weight.

A beginning could perhaps be made by giving one of the rocket manufacturers a contract for a mock stage of such a vehicle, which would have a complete electrical checkout and telemetry system installed. The thing could be factory built, flown to the Cape, trucked around the area, erected, returned to hangar, checked again, etc. If the new electronics is as good as people say, this mock stage would not have to be rebuilt between each mere movement, and a feeling for the concept could perhaps be obtained. If it works, we can then plant tulips in the cable trenches.

SPACECRAFT

The design of the propulsion and auxiliary equipment for the spacecraft is tricky. The 18,000 fps velocity increment must be delivered by a propellant combination storable in space covering the whole range from Mercury to Neptune. The stage weight/payload ratio for oxygen difluoride and diborane, the highest performance space storable propellants known to me, is shown in Figure 16. Assuming conservative structures because of atmospheric braking requirements, it still should be possible to carry roughly 500 pounds of actual payload in a 3,000 pound spacecraft.

Communications and Power Supply

The use of 25 watts power radiated from the spacecraft in conjunction with the 210 foot dishes and other improvements scheduled to be operational in the DSIF by 1967, would provide for the return of approximately 2000 data bits per second from Mars.[6] This would permit the transmission of modest quality (120,000 bits per picture) television pictures at a rate of one per minute from Mars, and a rate of two per day from Pluto. To my mind, this data rate would be perfectly reasonable. Any higher rate from Pluto should be by development of laser techniques, which give promise of reducing power requirements by a factor of about 100. The communication equipment should not weigh more than the 60 lb of Mariner II, and would require about 75 watts of power. A reasonable payload breakdown, then, would be 300 lb of scientific instruments and 60 lb of communication gear leaving 140 lb for the power supply.

The deep solar space missions rule out solar cells as a power source. A nuclear isotope battery would appear to be the most logical source of power, although nuclear reactors or combined supplies might have merit. Since about 210 watts would be required for both communication and experiments, the battery must weigh no more than 0.67 lb per watt. It must have the order of a ten year life. These requirements tend to dictate a strontium-90 battery, which requires the spacecraft designers to put up with the problems of β emitters. Plutonium-238 is expensive and only available in small amounts. Even with strontium-90, the provision of 24 batteries per year may tax the isotope supply, but the cost per battery should be only a few hundred thousand dollars.

It appears feasible to provide a communications system with current techniques, planned DSIF improvements, and a new nuclear isotope power supply which will adequately cover the solar system and permit perhaps 300 lb of scientific instrumentation in a 500 lb total payload package.

Guidance and Braking

The spacecraft, of course, has further tricky design features. It must have a navigation system which will permit it to go into orbit upon approach to other planets and on occasion rendezvous with and soft land upon their satellites. It must be able to perform atmospheric braking in the atmospheres of the four major planets, as well as Mars, Venus, and Mercury. I would expect that complaints will be raised that guidance for such operations would be too complicated, and that not enough is known to permit reliance upon atmospheric braking.

With the miniaturization possible in optical sensors, a relatively simple system of optical triggering of impulse by planetary and satellite images, particularly with programming transmitted from Earth subsequent to planetary capture, should solve the guidance problem with little weight and little impulse expended. In the latter connection it is interesting that although NASA originally allowed a few thousand fps impulse for guidance in Apollo system studies, the current guidance system, now that a reasonably refined study has been performed, only requires on the order of a hundred fps to do the job. Furthermore, as previously indicated, the complete payload velocity capability is only needed for a few missions, and early missions to selected targets could have large velocity margins available until refined guidance is checked out.

As for atmospheric braking, it seems that once again the aerodynamicists have work to do. The removal of only 20,000 fps high in the Jovian atmosphere may not be difficult, even at speeds of almost 200,000 fps. The probe would likely be subjected to atmospheric drag for about 5 minutes, and would thus experience an average deceleration of only 2 g's. The heat input should not be great. Furthermore, the corridor might not be too narrow since, although Jupiter's surface gravity is over twice Earth's, its large diameter gives a lower rate of decrease of gravity at higher altitudes. We need to expand our atmospheric studies to cover the removal of relatively small amounts of velocity upon first pass in the atmospheres of the major planets. These are meteoric speeds, but not meteoric energy inputs. It is not clear that the spacecraft envisioned is difficult to design after the initial shock wears off, but the multiplicity of design conditions does require some thought on the part of the spacecraft designers.

PROGRAM SIZE AND COST

In my opinion it is possible to justify the cost of developing such a vehicle and payload, even though NASA is rightfully leery of new vehicle developments. Such a vehicle is essentially three new high-energy stages, ranging in size from somewhat smaller than Centaur to somewhat larger than the Saturn SIV-B stage. Even if we were to assume a development program as beset with difficulties as Centaur, the price of the development of the new stages should not exceed $500 million each. The launch vehicle cost per shot after development, if cost effectiveness is made a paramount design objective, should be on the order of $10 million. Two dozen launches per year at $10 million per launch is only $250 million. Assuming that the payloads would also cost $10 million, only $500 million per year would enable 24 shots per year to be placed anywhere desired within the solar system. As previously indicated, supplying the nuclear isotope battery could be the limiting item on number of shots per year.

If we do it right, the entire solar system may be reached with an expenditure of approximately 50 percent of the current OSS yearly budget of about $1 billion, with the entry price a $1 to $2 billion (over 4 years) vehicle and payload development program. The current planning for Voyager covers only a miniscule portion of the missions of the program suggested here, but will require the same time scale. We should reorient Voyager to these concepts immediately, before another limited objective development is started.

A number of reports promoting the use of electrical rockets for space probe missions have concluded that a large fraction of solar missions cannot be performed with chemical rockets, References 7, 8. In my opinion, these reports err in not using efficient chemical rockets, using a Saturn I launch vehicle with the constraints it places on payload/velocity capabilities, or not considering the combined use of chemical rockets and planetary gravity fields. It appears to me that chemical rockets can perform all foreseeable missions, and at a far earlier time than current nuclear electric rocket developments. Hence, to the cost factors considered above should be added the potential cost savings of not developing unneeded electrical rockets.

The trucking base for scientific solar system exploration envisioned here could be expected to be so versatile that it would probably only be replaced at some future time either by the advent of manned stations throughout the solar system, or, prior to that time, by the utilization of gaseous fission spaceships to project the payloads at the required speed during the course of crew training missions for manned operations. In either event, we will probably not have such spaceships for at least 10 years, and they may not be available for scientific payload projection during the early years of their operation anyway due to required use in manned expeditions. Assuming a four-year development time for the system, its useful life should be on the order of at least six to ten years. Thus, close to a minimum of 200 rockets and payloads would be produced, which is enough to get most of the advantage from production-line techniques in the reliability and cost of vehicles, spacecraft and payloads as well as the cost of launching operations.

The number of 24 shots per year is somewhat arbitrary, but not completely so. I would expect those responsible to assign priorities to the missions and come up with a required number per year. Meanwhile my own rough numbers per year obtained by simply looking at the target complex, would be: 3 each to Mars and Venus, 2 each to Jupiter, Saturn, and the asteroids, and 1 each Solar probe, out-of-ecliptic, cometary, Mercury, Uranus, and Neptune or Pluto. I would assign the other 6 probes to engineering data-gathering missions to supply the information needed for the design of the manned ships to follow.

Scientific personnel should not be bothered with this engineering data requirement, yet it is important if we are not to have another data deficiency such as occurred when we had to design Apollo with practically no lunar surface information. It is not inconceivable to me that the Martian probe program could be run over by a Martian manned program if the Russians should by some remote chance beat us to the Moon, and we were to react violently. At least, it is no more inconceivable than the similar reaction to their manned orbital success in the spring of 1961. The analogies are, in fact, quite chilling.

CONCLUSIONS

The provision of a launch vehicle with 56,000 fps velocity capability would create the ability to place scientific payloads at any point in the solar system with travel times not to exceed seven years even to the deep planets. Such a vehicle would also open the launch windows to Mars and Venus to almost half a year. The use of close flybys of Jupiter in addition would permit 90° out-of-ecliptic and close solar probe missions, and would reduce the maximum travel time to above five years. The Soviets used a close flyby of the Moon for Lunik III trajectory control in 1958.

The provision of 18,000 fps velocity capability in the spacecraft coupled with clever guidance and atmospheric braking capability would permit landing on all of the satellites and planets of the solar system except the four inner satellites of Jupiter. The use of Callisto flybys, or the carrying of the third stage of the launch vehicle to Jupiter for extra braking impulse, represent possible means of reaching even those. Atmospheric braking should be possible at Mercury, and is needed at Pluto for landing missions.

The energy requirement for a 60,000 fps vehicle is only a little over twice that required of an escape vehicle. Since this is the amount of improvement achieved by high energy chemical propellants, a 450,000 pound, three stage hydrogen/fluorine launch vehicle with 3000 pound spacecraft should carry 500 pounds of payload for all missions.

It should be possible to obtain communication equipment and power supply adequate for the whole solar system with 200 pounds weight, so that 300 pounds of scientific instrumentation could be carried for these missions. The provision of a suitable nuclear isotope battery may be the limiting item on number of missions per year.

A communication relay planetoid should be established at one of the Trojan libration points of the Earth/Sun system to permit continuous communications as far in as Mercury. Laser techniques should be developed if very high data rates are needed from deep space.

The launching of 24 shots per year would provide a reasonable amount of solar exploration and create a high enough rate to effect production-line type savings in vehicle, spacecraft, and launch costs. Such a system should be used for at least 10 years since only the advent of large scale manned exploration or the use of gaseous fission powered spaceships to project scientific payloads, could economically replace it.

A strong effort for a modern, cost effective vehicle should be mounted. If only a small fraction of the theoretical cost savings can be achieved, it is reasonable to expect to be able to probe the entire solar system with a yearly expenditure rate of approximately 50 per cent of the current budgetary outlay of the NASA Office of Space Sciences.

The alleged requirement for nuclear electrical rockets for solar system probes is highly debatable. High energy chemical rockets can do all missions much sooner, and should receive the development funds.

Not only does it appear reasonable in terms of cost to open the whole solar system to scientific probes, but it would appear to be highly desirable scientifically. The Voyager program should not be pursued merely as a Mars/Venus program, but reoriented along the lines of this report. Otherwise, it will be another limited objective development.

A fertile field for cooperation with the Soviets is in the unmanned scientific exploration of the solar system. This report provides a rational basis for planning such future endeavors at earlier dates than commonly thought feasible.

Our current space science program spreads out from Earth in a geocentric manner curiously reminiscent of the religious view of the Universe prior to Copernicus and Galileo. This report takes the heliocentric view that the Good Lord did not put all worthwhile scientific phenomena, or even a significant portion of it on this particular planet. In ancient times, such attitudes have proved quite dangerous. I await the Inquisition.

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APPENDIX

SOLAR SYSTEM DATA

SOLAR BODY	Semi-Major Axis A. U.	Semi-Major Axis R⊙=1	Planet Mean Diameter Miles	Planet Mean Diameter ⊙=1	Mass Ratio ⊙=1	Surface Gravity ⊙=1	Period About Primary Days	Period About Primary Years	Orbit Vel. About Sun fps	Escape Velocity fps
SUN			869,000	101	333,500	27.7				2,020,000
MERCURY	.387		3,010	.38	.053	.367	88	.240	157,000	13,700
VENUS	.723		7,710	.97	.815	.862	225	.615	114,800	33,600
EARTH	1.00		7,920	1.00	1.00	1.00	365	1.00	97,600	36,700
Moon		60.27	2,160	.272	.0123	.166	27.32			7,800
MARS	1.524		4,220	.53	.107	.376	687	1.881	79,100	16,400
Phobos		2.775	10				.32			
Deimos		6.919	5				1.26			
ASTEROIDS										
Ceres	2.767		460				1681	4.61		
Pallas	2.767		300				1684	4.61		
Juno	2.670		120				1594	4.37		
Vesta	2.361		240				1325	3.63		
JUPITER	5.203		88,600	11.20	318.0	2.54	4333	11.86	42,800	196,000
V		2.539	100				.50			
Io		5.905	2,060	.26	.0132	.195	1.77			8,250
EUROPA		9.396	1,790	.23	.0080	.156	3.55			6,900
GANYMEDE		14.99	3,070	.39	.0256	.170	7.15			9,430
CALLISTO		26.36	2,910	.37	.0151	.112	16.69			7,450
VI		160.1	75				250.6			
VII		164.4	25				259.8			
X		164	12				260			
XII		290	12				625			
XI		313	15				696			
VIII		326	25				739			
IX		332	14				755			

APPENDIX (Con't.)
SOLAR SYSTEM DATA

SOLAR BODY	Semi-Major Axis A. U.	Semi-Major Axis R⊙=1	Planet Mean Diameter Miles	Planet Mean Diameter ⊙=1	Mass Ratio ⊙=1	Surface Gravity ⊙=1	Period About Primary Days	Period About Primary Years	Orbit Vel. About Sun fps	Escape Velocity fps
SATURN	9.546		75,000	9.47	95.22	1.06	10,759	29.46	31,600	116,000
Mimas		3.111	300				.94			
Enceladus		3.991	350				1.37			
Tethys		4.939	750	.09	.00011	.013	1.89			1,310
Dione		6.327	800	.10	.00017	.017	2.74			1,540
Rhea		8.835	1,100	.14	.00039	.020	4.52			1,950
Titan		20.48	3,100	.39	.0230	.150	15.95			8,900
Hyperon		24.83	250				21.28			
Japetus		59.67	750				79.33			
Phoebe		216.8	200				550			
URANUS	19.20		29,600	3.74	14.55	1.04	30,687	84.02	22,200	72,400
Miranda		5.494					1.41			
Ariel		8.079	350				2.52			
Ubriel		11.25	250				4.14			
Titania		18.46	600				8.71			
Oberon		24.69	500				13.46			
NEPTUNE	30.09		27,800	3.50	17.23	1.41	60,184	164.78	17,800	81,600
Triton		15.85	2,500	.31	.0252	.256	5.88			10,400
Nereid		249.5	200				500			
PLUTO	39.5		9,000?	1.1?	.9	.705	90,700	248.4	15,500	32,700

All data from Reference 9 except:

Satellite orbit semi-major axis from Reference 10.

Surface gravity and escape velocity calculated from diameter and mass raio.

FIGURE 1

SOLAR SYSTEM VELOCITY REQUIREMENTS

FLYBY MISSIONS OR
ATMOSPHERIC BRAKING AT ARRIVAL

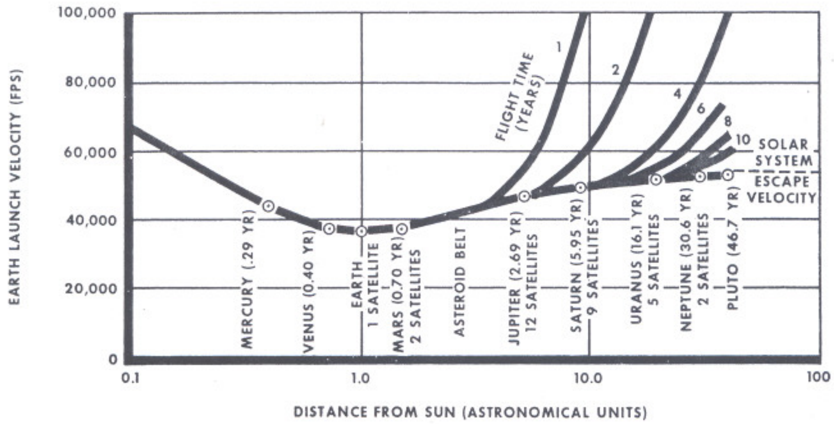

FIGURE 2

SYNODIC PERIOD OF PLANETS

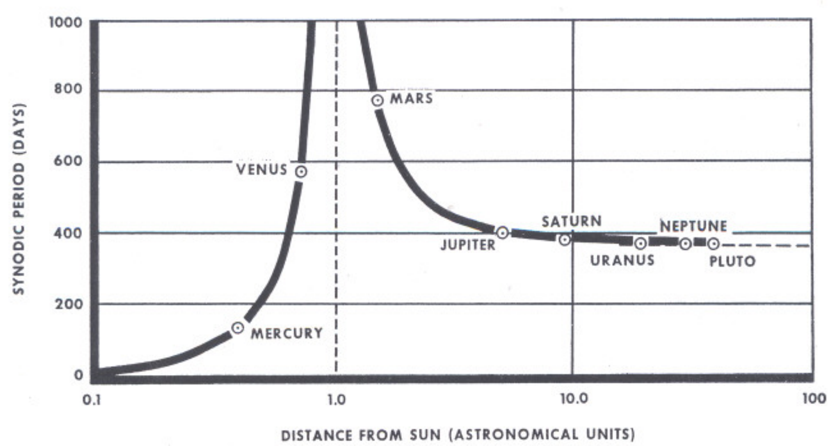

FIGURE 3

SOLAR SYSTEM VELOCITY REQUIREMENTS

COMPARISON OF SIMPLIFIED AND EXACT CALCULATIONS

LAUNCH WINDOWS
60 DAYS-MARS AND VENUS DIRECT
30 DAYS-ALL OTHERS

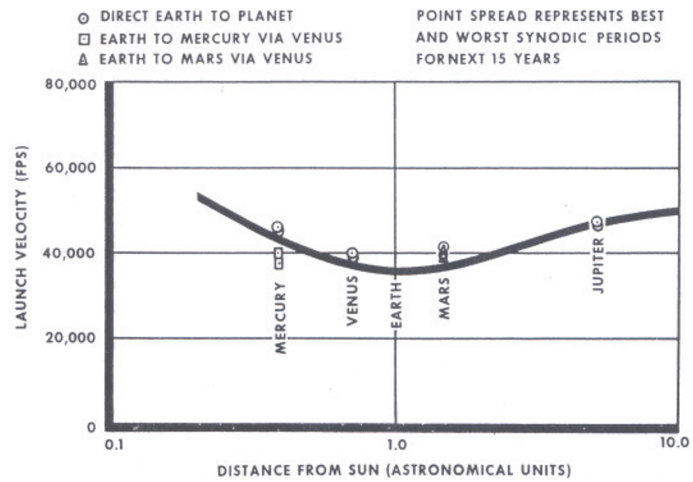

FIGURE 4

EARTH-MARS LAUNCH WINDOWS

ATMOSPHERIC BRAKING AT MARS
LAUNCH VELOCITY=60,000 FPS
CONTOURS WERE CALCULATED FOR
1964-65 LAUNCH PERIOD

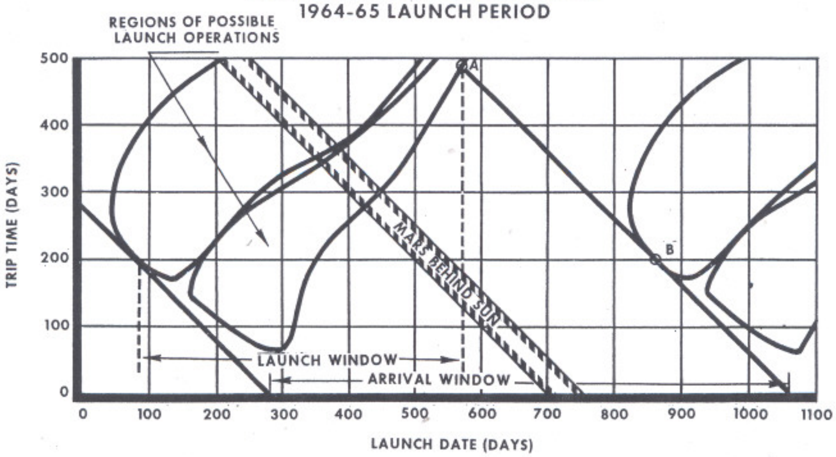

FIGURE 5

EARTH-MARS LAUNCH WINDOWS

LAUNCH VELOCITY=60,000 FPS.
ATMOSPHERIC BRAKING AT MARS

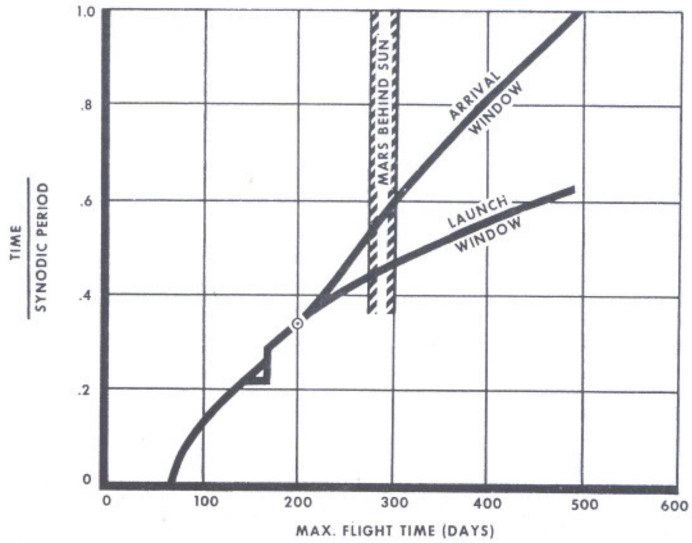

FIGURE 6

SOLAR PROBE VELOCITY REQUIREMENTS

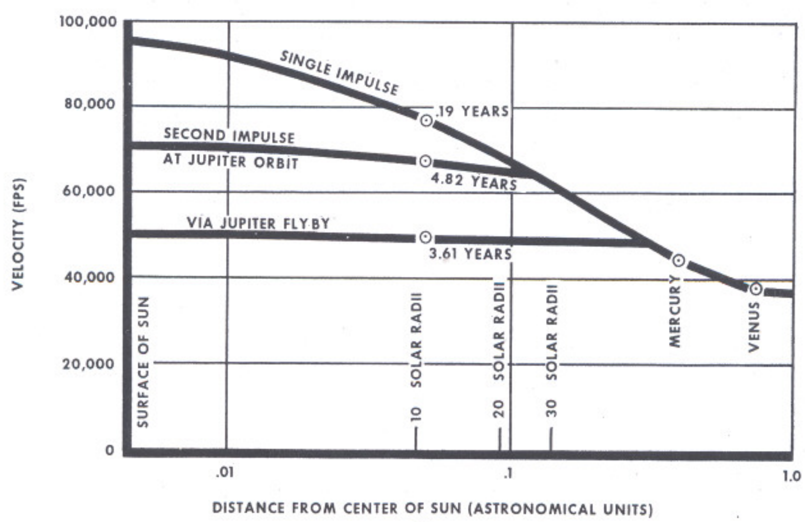

FIGURE 7

OUT-OF-ECLIPTIC VELOCITY REQUIREMENTS

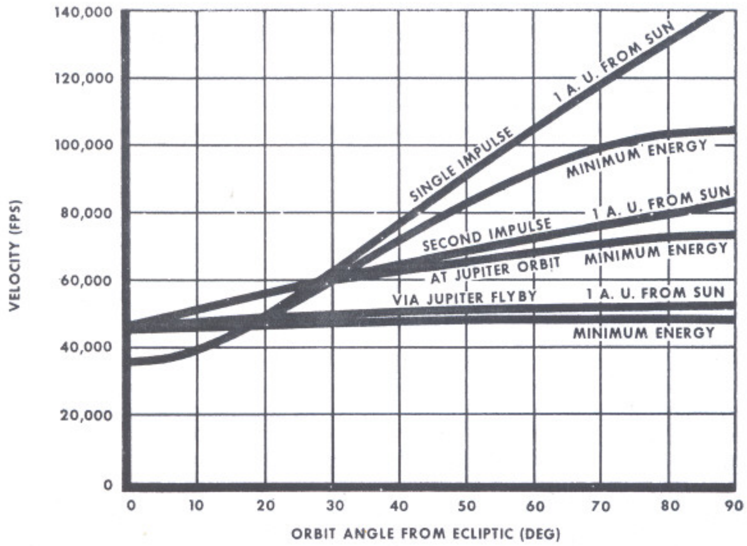

FIGURE 8

DEEP SPACE TRAVEL TIMES

VIA JUPITER FLYBY

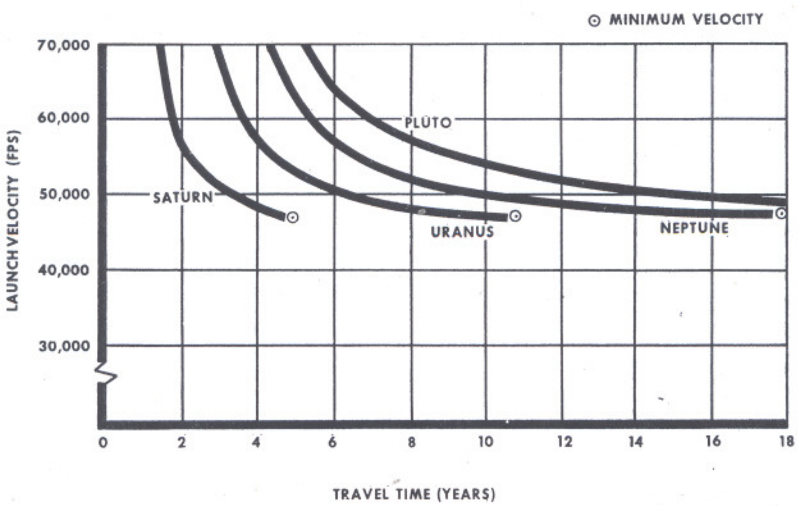

FIGURE 9

SOLAR SYSTEM VELOCITY REQUIREMENTS

BRAKING VELOCITY REQUIRED
TO MATCH PLANETARY ORBITS

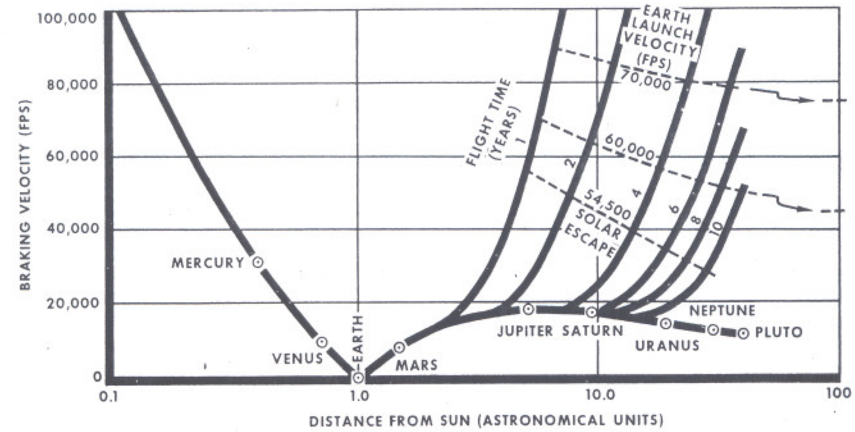

FIGURE 10

SOLAR SYSTEM VELOCITY REQUIREMENTS

COMPARISON OF SIMPLIFIED AND EXACT CALCULATIONS

LAUNCH WINDOWS
60 DAYS-MARS AND VENUS DIRECT
30 DAYS-ALLOTHERS

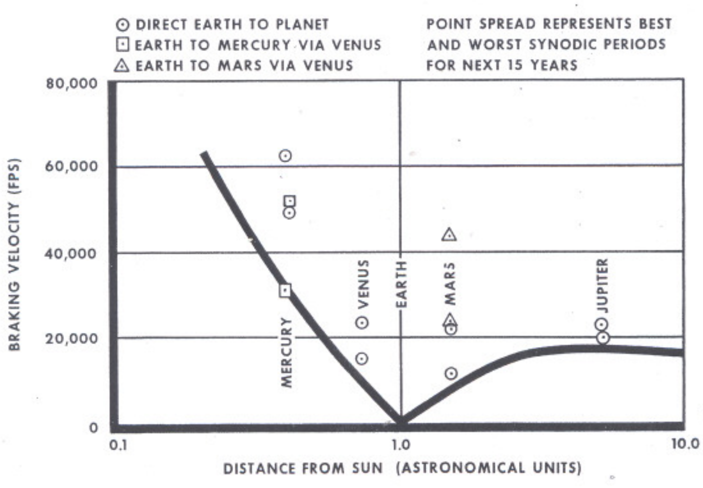

FIGURE 11

PLANETARY CAPTURE VELOCITIES

APO-APSIS OF CAPTURE ORBIT=100 PLANETARY RADII
BRAKING VELOCITY ASSUMED APPLIED AT PLANETARY SURFACE

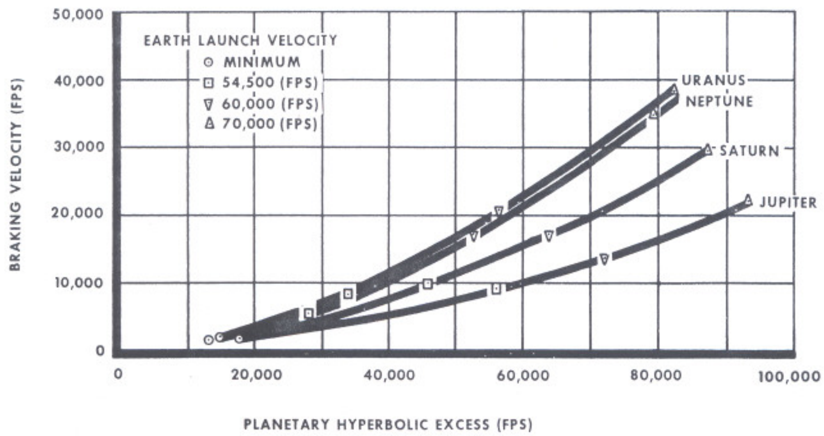

FIGURE 12

PAYLOAD VELOCITY REQUIREMENTS

LANDING ON SATELLITES OF MAJOR PLANETS
APO-APSIS OF CAPTURE ORBIT=100 PLANETARY RADII

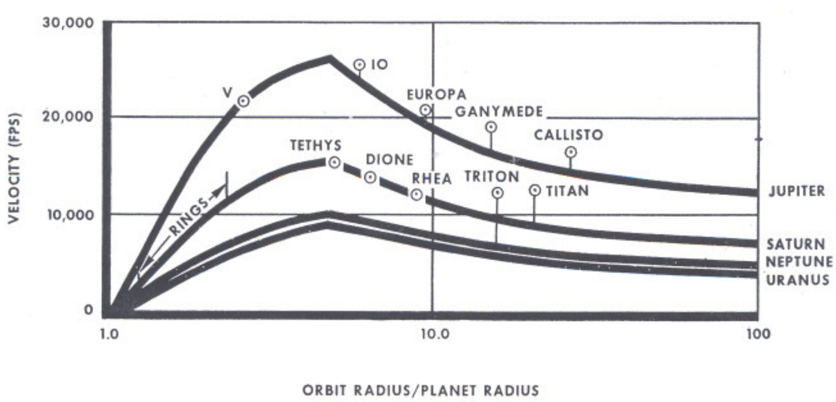

FIGURE 13

SOLAR SYSTEM BODIES WITHOUT ATMOSPHERES*
SURFACE GRAVITY vs ESCAPE VELOCITY

*ATMOSPHERES HAVE BEEN DETECTED ON MERCURY AND TITAN

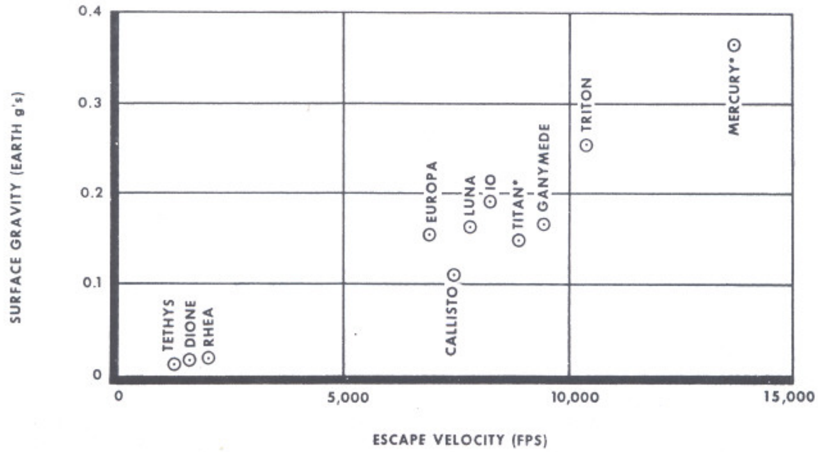

FIGURE 14

SOLAR SYSTEM BODIES WITHOUT ATMOSPHERES*
DIAMETER vs MASS

*ATMOSPHERES HAVE BEEN DETECTED ON MERCURY AND TITAN

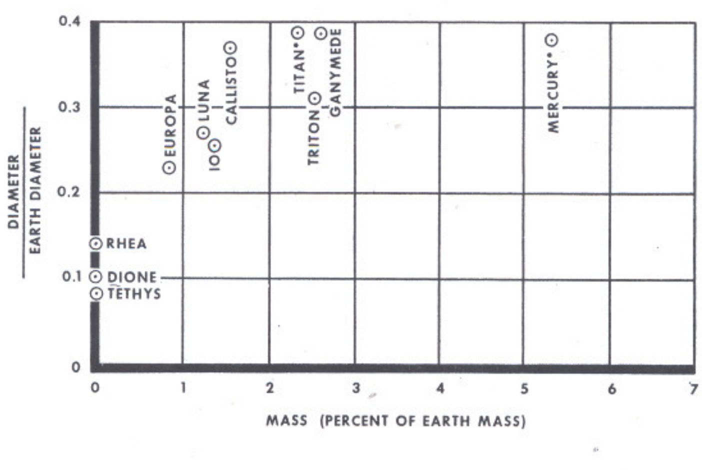

FIGURE 15

INITIAL WEIGHT/PAYLOAD WEIGHT

HYDROGEN/FLUORINE PROPELLANTS

FOR N - STAGE VEHICLES $V_{TOTAL} = N\,V_{STAGE}$ (isp=465 SEC)

$\left(\frac{W}{PL}\right)_{TOTAL} = \left(\frac{W}{PL}\right)^{N}_{STAGE}$

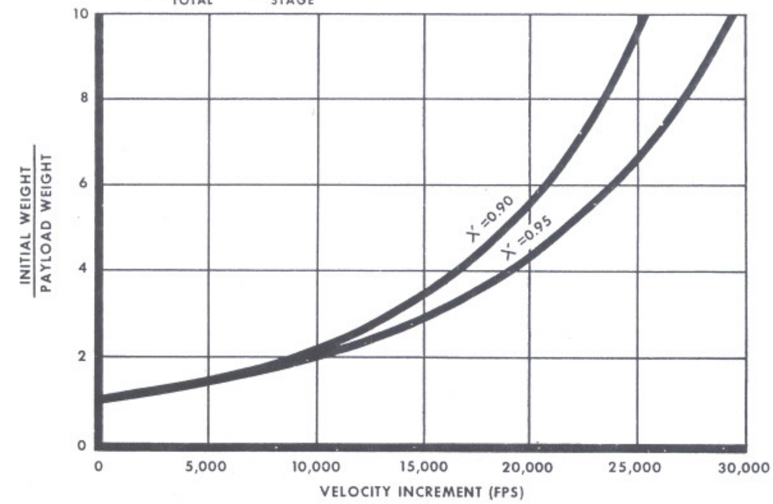

FIGURE 16

STAGE WEIGHT/PAYLOAD WEIGHT

SPACE STORABLE PROPELLANTS

OXYGEN DIFLUORIDE/DIBORANE

(isp=405 SEC)