Future unmanned exploration of the solar system

The whole solar system can be opened to scientific space exploration by developing a single modern chemical rocket vehicle and versatile payload and communication package specifically for this purpose

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This article presents a more comprehensive program for unmanned solar-system exploration than now planned. Certain data come from recent studies, and some suggestions are already familiar to vehicle designers. Other information and viewpoints are my own, and need refinement by detailed study.

Why consider a more comprehensive unmanned lunar and planetary program? The current national effort aims almost exclusively at the Moon and the planets Mars and Venus. Actually there are eight planets besides Earth, 30 known natural satellites besides 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-directions, as advancing technology, with or without competition, inevitably raises questions of program adequacy. It could lead, moreover, to immature approaches if the U.S. should establish cooperative planetary probe programs with the Soviet Union without understanding the fundamental factors involved.

There has existed 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 further than the Moon, Mars, and Venus, at least for a long time to come. The argument to be presented attacks this assumption. The solar system as an entity represents one of the few natural "space program package plans" available. And it does not unduly stretch the imagination. The distance to the nearest star is 7000 times the distance to Pluto, but Pluto is only about 26 times the distance to 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 would mean 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 bodies with atmospheres. Rocket impulse would be required for course changes and guidance corrections, and for braking into orbits. Furthermore, it would be extremely desirable to use atmospheric braking to enter an orbit and so 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 anywhere in the solar system. This will obviously require higher velocities than our current probes attain. Three separate aspects of launch velocity will be considered: The ability to reach anywhere in the solar system, including excess velocity to reduce travel time to distant targets; excess velocity to open the launch windows to Mars and Venus, so that much closer to year-round operations would be possible; and out-of-ecliptic and solar-probe missions. Normal trajectories will be considered first. Some effects of unconventional trajectories making use of the energy available from planetary gravitational fields will be examined, as well as payload velocity requirements to establish orbits about planets and to land on satellites.

The discussion assumes circular coplanar orbits for all planets and satellites. A satellite orbit is 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 for planets to indicate the validity of the approach. The table on page 18 gives the pertinent characteristics of the planets and satellites in the solar system used here. Significant uncertainties exist in some of these quantities, but they should not be great enough to change the basic conclusions.

Launch Velocity. Top left graph on page 17 shows the basic launch velocity required. The lower curve gives the launch velocity to reach the various planets with minimum energy, together with travel time for this condition and curves of the velocities re-

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quired to reduce flight time to the further planets. As an example, minimum-energy flight to the planet Uranus requires 52,000-fps launch velocity and takes 16.1 yr; but this could be reduced to 4 yr by using 63,000-fps launch velocity.

It is desirable to reduce travel time to the far planets for several reasons, reliability of equipment being perhaps the most obvious. For example, start to think of flight times of 16 yr (or 47 yr to Pluto) and you must weigh a state-of-the-art vehicle development and travel time against whatever new system will eventually replace it. This probably restricts maximum flight time to the order of 10 yr, as will be discussed later. (I shall refrain from discussing the importance of the political synodic period, which obviously has a major period of 4 yr 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 move 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 appears in the graph on page 18.

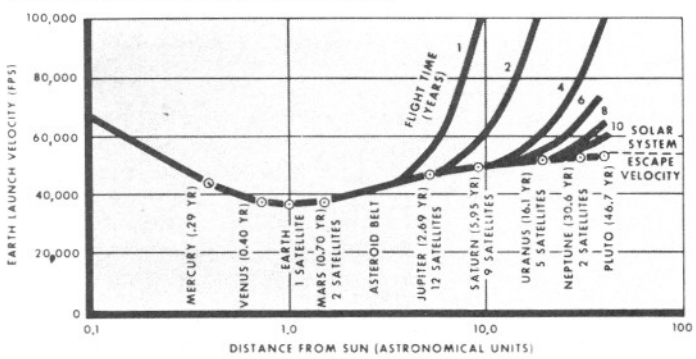
The top right graph below summarizes the effects on launch velocity of 60-day launch windows to Mars and Venus.[1] These data represent accurate machine calculations, including actual planetary orbital eccentricities and inclinations, as compared to the simplifications presented in the first graph. The highest and lowest velocities shown for each pair of symbols represent respectively the worst and best synodic periods during the next 15 yr. Only about 8% 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 Mars will be discussed to illustrate the phenomena involved. The contours shown in the top left graph on page 19 represent 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 simply be storing the probe in space, rather than on Earth. It can be seen that, although a completely open arrival window is available, launching should occur only roughly half of the time.

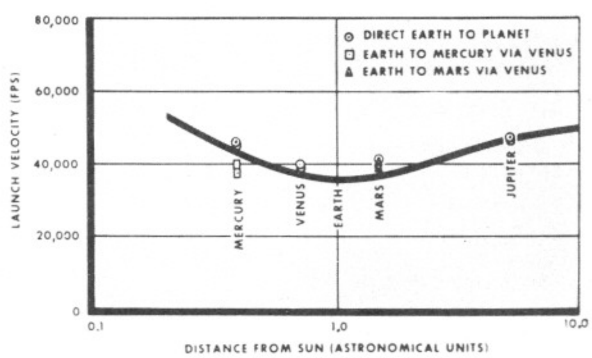
Additional contraints are evident. A completely open arrival window requires a maximum flight time of 490 days, and one might arbitrarily decide to limit this value to some smaller number. If so, both arrival and launch windows will be correspondingly reduced. The top right graph on page 19 plots both launch and arrival windows versus maximum flight time.

At least one other limitation exists. Making use of the completely open arrival window puts Mars part of the

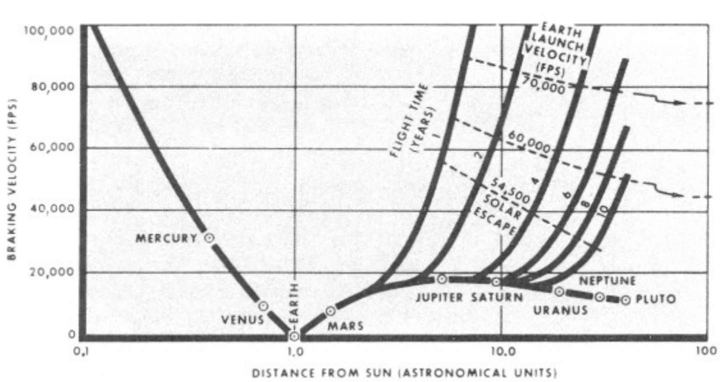
SOLAR-SYSTEM VELOCITY REQUIREMENTS



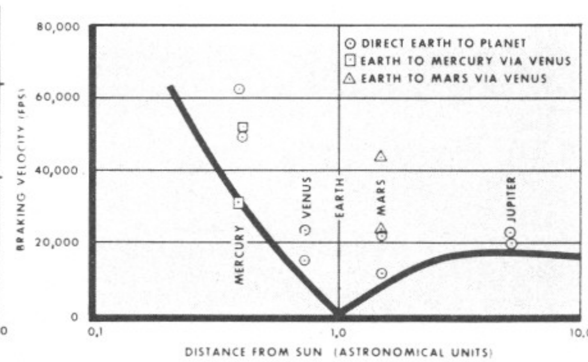
A. **Flyby missions or atmospheric** braking at arrival.



B. **Launch-velocity comparison,** simplified and exact calculations. Launch windows: 60 days for Mars and Venus direct and 30 days for all others. Point spread represents best and worst synodic periods for next 15 yr.



C. **Braking velocity required** to match planetary orbits.

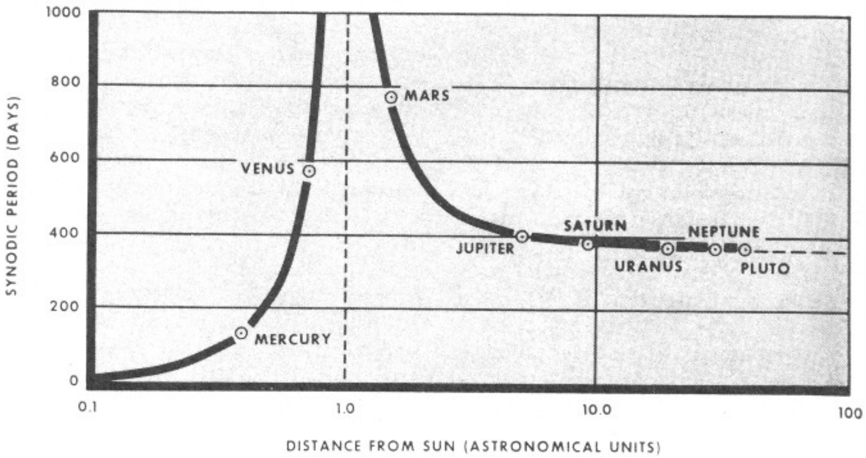


D. **Braking-velocity comparison,** simplified and exact calculations. Launch windows: 60 days for Mars and Venus direct and 30 days for all others. Point spread represents best and worst synodic periods for next 15 yr.

time on the opposite side of the Sun from the Earth, where the spacecraft either would not be able to communicate with it and would have to store data for later transmission, or would have to 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. Just 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. The band of time during which Mars is hidden from the Earth by the Sun can be seen in the two graphs at the top of page 19. 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 drop approximately 10% for propulsive rather than atmospheric braking at Mars.

SYNODIC PERIOD OF PLANETS



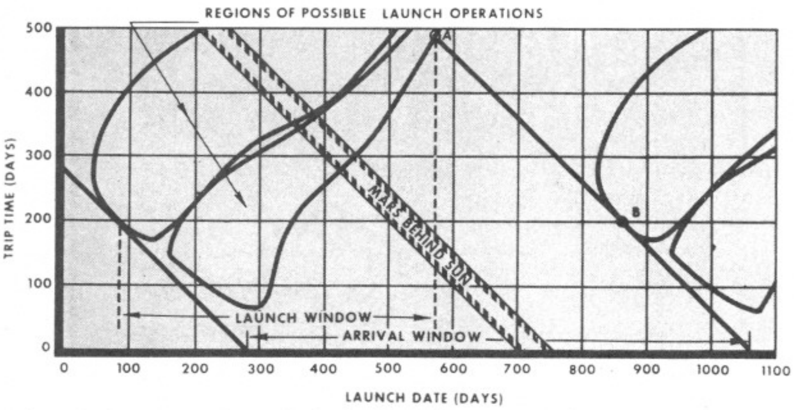
A considerable investigation of high launch velocities for both Mars and Venus would have to be made for a number of different synodic periods before these requirements can be pinned down accurately. It does appear, however, that 60,000-fps launch velocity will permit operation to Mars at least 40% of 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, launch velocities as high as 60,000 fps can vastly alleviate the launch-window

SOLAR-SYSTEM DATA

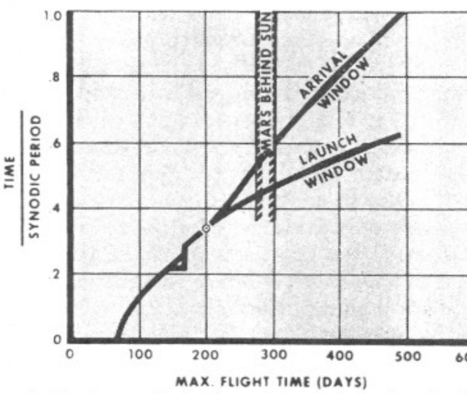
Solar body[a]	Semi-Major Axis		Mean diameter		Mass ratio	Surface gravity	Period about primary		Helio-centric velocity, fps	Escape velocity, fps
	A.U.	R(⊙ = 1)	Miles	⊙ = 1	⊙ = 1	⊙ = 1	Days	Years		
Sun	—	—	869,000	101	333,500	27.7	—	—	—	2,020,000
Mercury	0.387	—	3,010	0.38	0.053	0.367	88	0.240	157,000	13,700
Venus	0.723	—	7,710	0.97	0.815	0.862	225	0.615	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	0.272	0.0123	0.166	27.32	—	—	7,800
Mars	1.524	—	4,220	0.53	0.107	0.376	687	1.881	79,100	16,400
Phobos	—	2.775	10	—	—	—	0.32	—	—	—
Deimos	—	6.919	5	—	—	—	1.26	—	—	—
Asteroids	—	—	—	—	—	—	—	—	—	—
Ceres	2.767	—	460	—	—	—	1,681	4.61	—	—
Pallas	2.767	—	300	—	—	—	1,684	4.61	—	—
Juno	2.670	—	120	—	—	—	1,594	4.37	—	—
Vesta	2.361	—	240	—	—	—	1,325	3.63	—	—
Jupiter	5.203	—	88,600	11.20	318.0	2.54	4,333	11.86	42,800	196,000
V	—	2.539	100	—	—	—	0.50	—	—	—
Io	—	5.905	2,060	0.26	0.0132	0.195	1.77	—	—	8,250
Europa	—	9:396	1,790	0.23	0.0080	0.156	3.55	—	—	6,900
Ganymede	—	14.99	3,070	0.39	0.0255	0.170	7.15	—	—	9,430
Callisto	—	26.36	2,910	0.37	0.0151	0.112	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	—	—	—
Saturn	9.546	—	75,000	9.47	95.22	1.06	10,759	29.46	31,600	116,000
Mimas	—	3.111	300	—	—	—	0.94	—	—	—
Enceladus	—	3.991	350	—	—	—	1.37	—	—	—
Tethys	—	4.939	750	0.09	0.00011	0.013	1.89	—	—	1,310
Dione	—	6.327	800	0.10	0.00017	0.017	2.74	—	—	1,540
Rhea	—	8.835	1,100	0.14	0.00039	0.020	4.52	—	—	1,950
Titan	—	20.48	3,100	0.39	0.0230	0.150	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	0.31	0.0252	0.256	5.88	—	—	10,400
Nereid	—	249.5	200	—	—	—	500	—	—	—
Pluto	39.5	—	9,000?	1.1?	0.9	0.705	90,700	248.4	15,500	32,700

[a]All data from Ref. 9 except satellite orbit semi-major axis from Ref. 10. Surface gravity and escape velocity calculated from diameter and mass ratio.

EARTH–MARS LAUNCH WINDOWS



A. **Launch-date picture.** Launch velocity, 60,000 fps; atmospheric braking at Mars; contours calculated for 1964–65 launch period.



B. **Maximum-flight-time** picture. Launch velocity, 60,000 fps; atmospheric braking at Mars.

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. Top right graph on page 17 shows an example of this for Earth–Mercury and Earth–Mars missions utilizing a close flyby of Venus for orbit modification.[3] Although the velocity requirements for Earth–Mars operations were not decreased significantly, it was possible to find Earth–Mars launch windows for half of the Earth–Venus launch windows investigated. Since Venus launch windows are more frequent than those of Mars, and rarely occur at the same time, this represents almost a doubling of available launch windows to Mars.

For flight to Mercury, it is possible to reduce the velocity requirements by about 4000 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 a third 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 deg 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 deg, adequate margin is available.

A further consequence of Jupiter's strong 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 1-deg change in deflected angle.

Besides the beneficial effects of strong gravity field, Jupiter comes close enough to the Sun that the increases in flight time for the Jupiter-flyby trajectories are not excessive.

The middle graph on page 20 denotes the use of a Jupiter flyby for solar-probe missions. The velocity required to come as close to the Sun as desired is only 50,000 fps if about 3 1/2 years of 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 this graph.

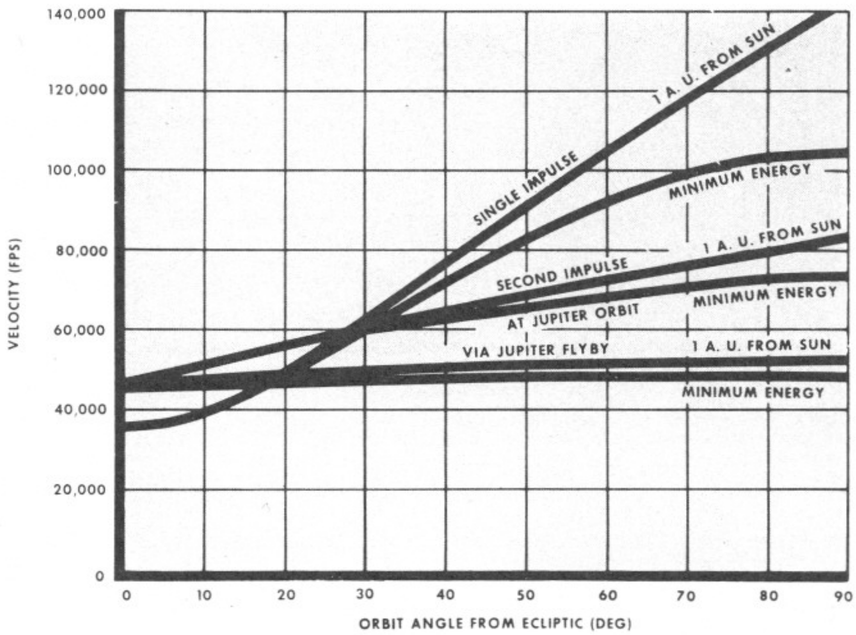
An even more startling flyby boost occurs in out-of-ecliptic trajectories, as shown in the top graph on page 20. To launch directly from the Earth 90 deg out of ecliptic and go over, or under, the Sun with a closest approach of 1 A.U. requires 140,000 fps. The same maneuver making use of a Jupiter flyby requires only 52,000 fps. Likewise, the requirement for going 90 deg out of ecliptic and making the closest possible approach to the Sun drops from 105,000 to 50,000 fps. The minimum-energy curves in the graph just cited represent varying degrees of close 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 deg, 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.

As the bottom graph on page 20 shows, the Jupiter gravity field can also be used to advantage for deep-space missions beyond this planet. 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. The flight time to Pluto using this lower velocity requirement is only 25 1/2 yr, compared to 47 yr without the aid of Jupiter's gravity field. The synodic periods of other outer planets and Jupiter create large intervals between launch windows that curtail the usefulness of this approach somewhat. The Jupiter–Saturn synodic period is almost 20 yr, and the other outer planet periods are about 13 yr. For solar-probe and out-of-ecliptic missions, however, the launch windows occur each Earth–Jupiter synodic period of just over one year.

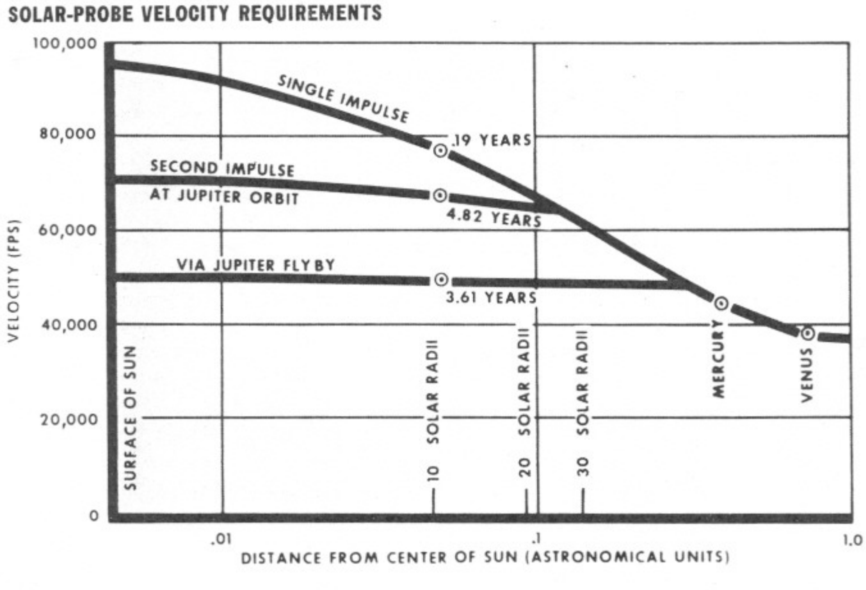
It should be realized that the effects of using planetary gravity fields are large, not merely minor perturbations. Particularly in solar-probe and out-of-ecliptic missions, velocity requirements decrease by roughly factors of two and come nicely into the range of other solar-system requirements. This 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 vehicles of needlessly high performance.

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

OUT-OF-ECLIPTIC VELOCITY REQUIREMENTS



SOLAR-PROBE VELOCITY REQUIREMENTS



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.

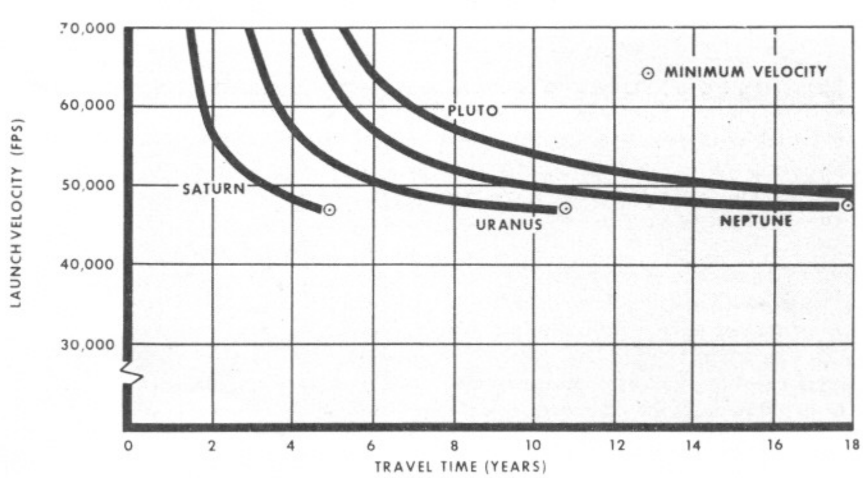
Newly discovered comets are quite a different matter.[5] In that case, velocity requirements to intercept the comet after detection but prior to solar passage could become extremely large.

Ruling 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. The first graph, on page 17, 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 can be seen in the bottom graphs on page 17, which give the braking velocities required for the launch velocities of the two graphs at the top of page 17. It can be seen that these curves increase very rapidly

TRAVEL TIME VIA JUPITER FLYBY



at Earth launch velocities beyond the minimum. This is due to the fact that high-speed rockets do not lose as much velocity 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. It also explains why bottom right graph on page 17 shows a much larger spread between the approximate curves and the actual calculations than in the launch velocities of the graph at top right of page 17. 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, being farthest away, with a small gravity field and no known atmosphere, 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. 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 closest 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. Then, 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, at 6 planetary radii. Establishing 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 appear in the lower graphs on page 17. The Mercury requirement is large because of the high orbital speeds close to the Sun and the planet's high inclination. The Pluto requirement is large if high launch velocities are used to decrease travel time. 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 ft. 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-ft altitude. Pluto is thought to be slightly larger than Earth, and might have an atmosphere undetected due to the great distance of observation. Atmospheric capture at Pluto would be possible only if it has an atmosphere 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 there is not confidence enough in the use of atmospheric braking. The first probe could 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. Indeed, atmospheric braking might well be used at an early date. What is already there should be applied when it comes to flight mechanics; 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 makes maximum use of planetary atmospheres and gravity fields to reduce propulsive velocity requirements to a minimum. First nearing 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 apoapsis (point farthest from planet-apogee at Earth) of this orbit will be limited to about 100 planetary radii. This will result in an orbital period of less than one month for the major planets. Top graph on page 23 shows capture requirements for the major planets. Evident is the very strong attenuation of the braking requirements given in the bottom graphs on page 17 by the planetary gravity fields.

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 moving at high speeds, and without exact knowledge of atmospheric 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, as indicated in the graph at top of page 23.

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 5 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 5 planetary radii, the probe would add an amount of velocity at initial apo-apsis to raise the periapsis 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 the lower graph on page 23.

Landing on the Satellites. The solar-system bodies without appreciable atmospheres, as seen through the eyes of a propulsive system designer, are represented at the top left on page 24—

a plot of escape velocity versus surface gravity. An idea of the size of these bodies may be obtained from the diameters given in top right graph on page 24. Ganymede and Titan are actually larger than Mercury, although Mercury has the largest escape velocity (13,700 fps). An atmosphere has been detected on Titan, but it is probably too small to be helpful.

A group of satellites similar to the Moon exist at 6900–10,400-fps escape velocity: Io, Europa, Ganymede, and Callisto—the four large satellites of Jupiter (the Galilean satellites); 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 only about 25% of those of the satellites just mentioned. 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 the 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 given in the lower graph on page 23, under the assumption that the orbit-matching maneuver takes place close 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. Some 18,000 fps would allow reaching Callisto. It might even be possible that Io, Europa, Ganymede, and Jupiter V could be reached within 18,000-fps spacecraft velocity by 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.

What Launch Vehicle? The previous sections outline a wide variety of mission possibilities, and it is possible to synthesize a number of different launch vehicle and spacecraft combinations to meet a significant fraction of these missions. Only one 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 time of the order of 5 yr would be 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 spacecraft propulsive velocity of 26,000 fps and Earth launch velocity of perhaps 48,000 fps. The 48,000 fps was selected to reduce somewhat 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 yr at minimum velocity. Requiring 26,000 fps in the spacecraft leads to a complicated design with 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 1500 fps beyond the theoretical minimum for landing on Callisto shown in the bottom graph on page 23, and should be an adequate allowance for guidance and control corrections.

The satellites of Jupiter within the orbit of Callisto—Ganymede, Europa, Io, and Jupiter V—can then be reached in one of two ways. The first involves 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 the planet 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 8000 fps must be provided so that the basic launch-vehicle speed must be 56,000 fps. Thus the vehicle postulated has a 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 if feasible.

CRITICAL LAUNCH MISSIONS

Mission	Velocity required, fps: Total	From payload	Launch vehicle
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	zero	48,000
Neptune/Triton[a]	57,000	6,000	51,000
Pluto flyby[a]	64,000	18,000	46,000

[a] 6 yr via Jupiter; approximately 9 yr direct.

DEEP-SPACE FLIGHT-TIME OPTIONS

Mission	Velocity, fps: From payload	From Launch	Flight time, yr: Direct	Via Jupiter
Jupiter/Io	−8000	48,000	1.8	—
Jupiter/Callisto	zero	56,000	1.0	—
Saturn/Titan	6000	62,000	1.8	1.7
Uranus/Miranda	8000	64,000	4.0	3.3
Neptune/Triton	6000	62,000	7.0	5.1
Pluto Flyby	18,000	74,000	6.6	5.2

The top table at the left shows the critical launch missions, with an arbitrary 6-yr flight-time limit and with the use of the Jupiter flybys for solar-probe, out-of-ecliptic, and deep-space missions. The bottom table at the left gives flight times for various deep-space missions for the postulated vehicle capability.

Vehicle Size. For such 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 graph on page 24 gives stage velocities achievable with hydrogen/fluorine rockets. With a stage of average structural efficiency, it is possible to generate about 27,000 fps with a launch/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.

The curves on this graph can give 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 3000-lb spacecraft were carried, the launch weight of the vehicle would be 450,000 lb, about the same as Titan II.

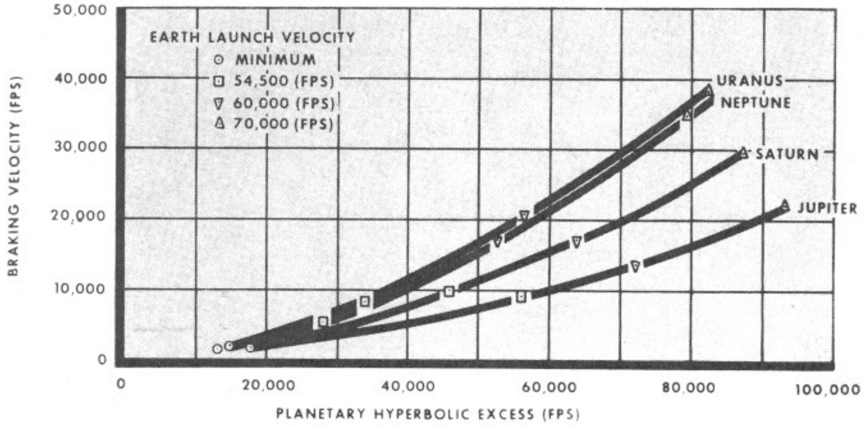
The table shown on page 23 gives a breakdown of stage weights and velocity increments. The spacecraft will be discussed later. If the third stage were usable for auxiliary spacecraft braking, it might be very difficult to achieve a λ' of 0.92 in this stage. Of course, if such a stage were also used on current vehicles, it might have to be designed for micrometeorite and long-term hydrogen storage requirements for Earth orbital missions. The effects of such requirements will not be considered here.

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

High-Energy Vehicles Compared

PLANETARY-CAPTURE VELOCITIES

Apo-apsis of capture orbit equals 100 planetary radii; braking velocity assumed applied at planetary surface.



HYDROGEN/FLUORINE ROCKET PROFILE

Stage	Stage wt., lb	Total wt., lb	Velocity Δ, fps	Stage ratio[a]
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[b]	6.0

[a] Initial weight to final weight.

[b] Assumes 6000-fps drag, gravity, and nozzle losses. Impulsive velocity = 22,000 fps.

with Conventional. This discussion purposely emphasized velocity requirements before considering vehicle design. This was done because of a feeling that too much current thinking concerns computer calculations that make no attempt to understand the interaction of rocket performance and flight mechanics. 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 we need, and the extra energy already exists in high-energy propellants.

As another example, it has been often stated that the launch-window problems of Mars and Venus present an insurmountable fact of Nature. Yet the "fact" usually quoted to prove this consists of payload versus launch date curves for Atlas-Agena—a marginal vehicle indeed for these missions.

Another approach[1,3] presents the energies to perform various missions from 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 to go from 40,000- 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 wastes 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 high speeds reasonably.

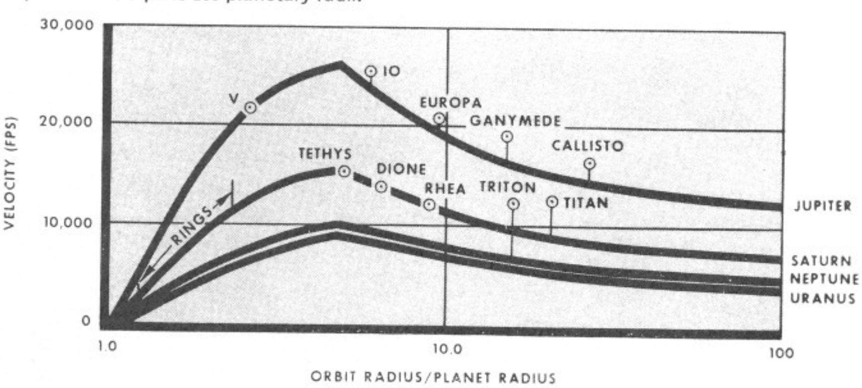
Some interesting analogies exist between the use of high-velocity rockets and the design of long-range transport aircraft. A long-range transport carries a great deal of fuel but only a small passenger or freight fuselage. It ignores the fact that at short ranges it is theoretically capable of carrying 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. 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 cost 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 described by the table shown at above left here can be replaced by a 4200 lb payload package. The two lower stages by themselves can place about 35,000 lb 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

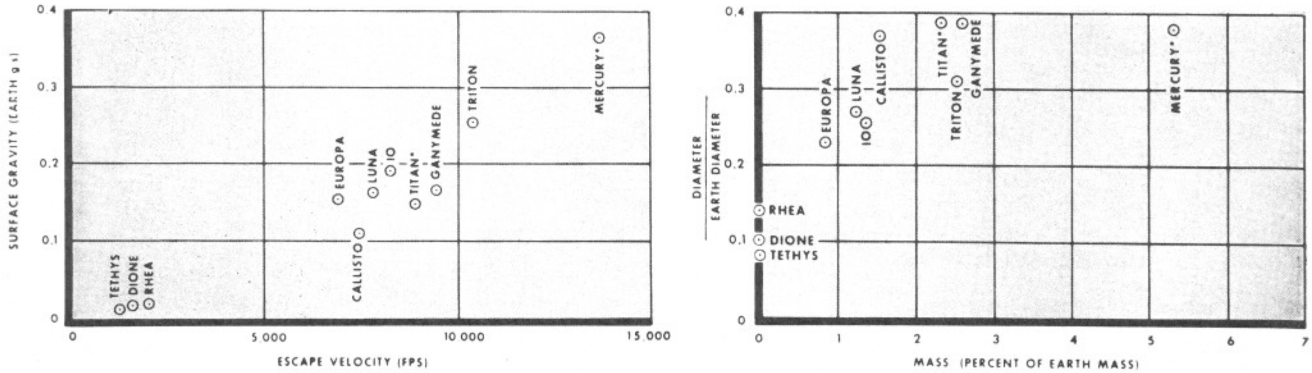
PAYLOAD VELOCITY REQUIREMENTS

Landing on satellites of major planets; Apo-apsis of capture orbit equals 100 planetary radii.



SOLAR-SYSTEM BODIES WITHOUT ATMOSPHERES

Atmospheres have been detected on Mercury and Titan.



A. Surface gravity vs. escape velocity.

B. Diameter vs. mass.

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, for instance, it should be possible to vacuum deposit the checkout instruments in the rocket with negligible weight penalty and so cut the ground-support facilities to the minimum.

Although this may seem a radical suggestion, I am convinced that no serious attempt really has yet been made to 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 components can be improved. This 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 inflight 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 that 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 are as good as reported, 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 plant tulips in the cable trenches.

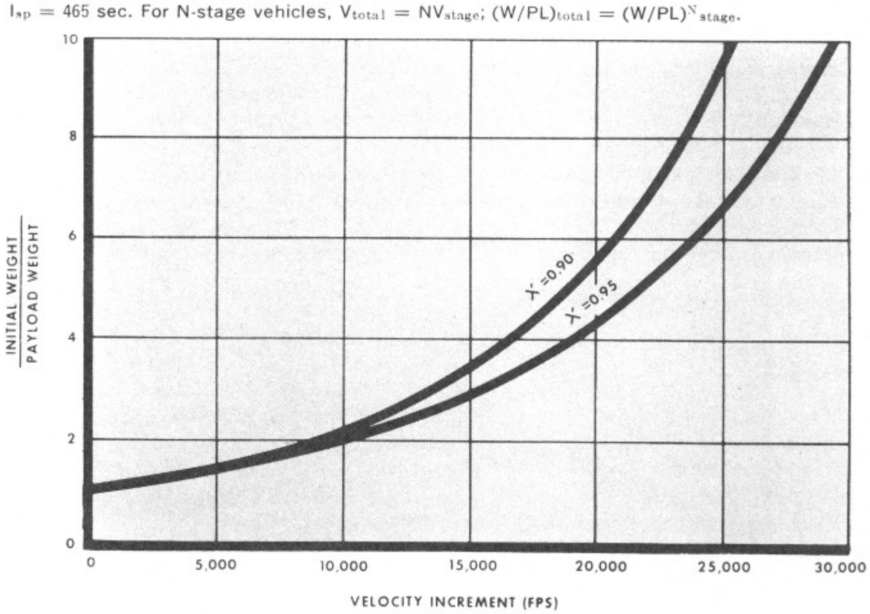
Spacecraft. The design of the propulsion and auxiliary equipment for the spacecraft would not be easy. The 18,000-fps velocity increment must be delivered by a propellant combination storable in space covering the whole range from Mercury to Neptune. The stage weight–payload ratio for oxygen difluoride and diborane—the highest performance space-storable propellants known to me—is shown in the graph on page 25. Assuming conservative structures because of atmospheric braking requirements, it still should be possible to carry roughly 500 lb of actual payload in a 3000-lb spacecraft.

Communications, Scientific Instruments, and Power Supply. The use of 25 watts of power radiated from the spacecraft in conjunction with the 210-ft dishes and other improvements scheduled to be operational in the NASA Deep Space Instrumentation Facility (DSIF) by 1967 would allow return of approximately 2000 data bits per second from Mars.[6] This would permit the transmission of TV pictures of modest quality (120,000 bits per picture) at a rate of one per minute from Mars and at 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 the Mariner II communication gear, and would require about 75-w power.

In current deep-space-payload technology, the weight of structure, wiring, and thermal control varies from 20 to 35% of total payload weight. The weight of attitude control varies from 10 to 15% of total payload

VEHICLE SIZE WITH H_2/F_2 PROPELLANTS

I_{sp} = 465 sec. For N-stage vehicles, $V_{total} = NV_{stage}$; $(W/PL)_{total} = (W/PL)^{N}_{stage}$.



weight. Assuming that a new payload utilizing latest techniques would achieve the lowest of these numbers, a total of 30%, or 150 lb, must be allotted to structure, wiring, and control. Assuming the specific weight of an isotope battery to be 0.67 lb per electrical watt, the 75 watts required for communication would require 50 lb of battery. Hence, a total of 260 lb would be required for structure, communication, and attitude control, leaving 240 would be 98 lb with 147-w output.

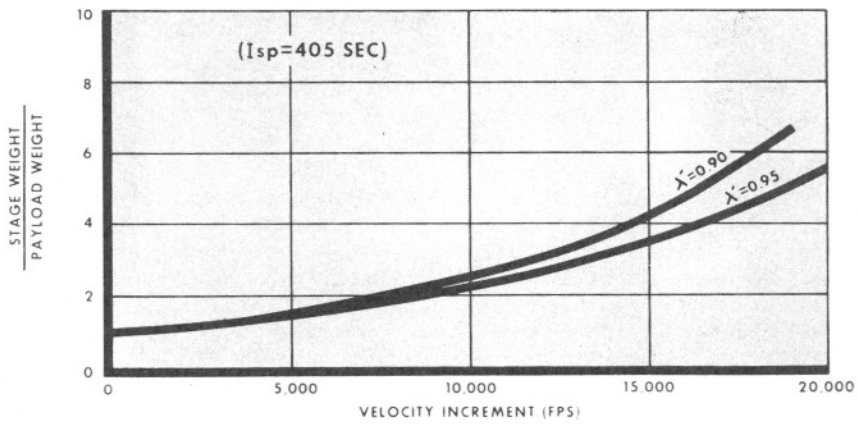
Power requirements average about 0.5 watt per pound of instruments. Hence, the 240 lb would consist of 180 lb of instruments and 60 lb of instrument battery. The total battery weight would thus be 110 lb with a total power output of 165 watts. If a more conservative structure and control allowance of 40% (200 lb) is necessary, then 142 lb of instruments would be carried and the total battery weight would be 98 lb with 147-w output.

The deep-space missions rule out solar cells as a power source. The nuclear-isotope battery would appear to be the logical source of power, although nuclear reactors or combined supplies might have merit. The battery must have the order of 10-yr life. This tends to dictate a strontium-90 battery, which requires the spacecraft designers to put up with the problems of beta 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 should be possible to achieve the specific weight of 0.67 lb per electrical watt assumed above with a strontium-90 battery of the size required.

It appears feasible to provide a communications system with current techniques, planned DSIF improvements, and a new nuclear isotope power supply that will adequately cover the solar system and permit over 150 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.

OXYGEN DIFLUORIDE/DIBORANE PERFORMANCE



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 a few thousand fps impulse was originally allowed for guidance in Apollo system studies, the current system, now that a reasonably refined study has been performed, only requires on the order of 100 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 min, and would thus experience an average deceleration of only 2 *g*. 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.

Program Size and Cost. In my opinion it is possible to justify the cost of developing such a vehicle and payload, even though we are 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 one assumed a development program as beset with difficulties as Centaur, the price of developing 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 the payloads would also cost $10 million, only $500 million per year would enable 24 shots per year to be directed to 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 a yearly expenditure of approximately 50% of the current space-science budget of about $1 billion. An initial expenditure of $1–2 billion, over a 4-yr period, for vehicle and payload development would be required. The current planning for future solar-system exploration covers only a miniscule portion of the missions of the program suggested here, but will require the same time scale. We should reorient these studies immediately, before another limited-objective development is started.

Some reports promoting the use of electrical rockets for space-probe missions have concluded that a large fraction of solar-system missions cannot be performed with chemical rockets.[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 and velocity capabilities, or not considering the combined use of chemical rockets and planetary gravity fields. Chemical rockets can perform all foreseeable missions, and at a far earlier time than current nuclear-electric rocket developments. Hence, to the cost factors cited 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, before then, by the utilization of gas-core nuclear-rocket-powered spaceships to project the payloads at the required speed during the course of crew-training missions for manned operations. In either event, it is not likely such spaceships will come into being for at least 10 yr, and they may not be available for scientific-payload projection during the early years of operation anyway, due to required use in manned expeditions.

Assuming a 4-yr development time for the unmanned system, its useful life should be on the order of at least 6–10 yr. Thus, close to a minimum of 200 rockets and payloads would be produced—enough to get most 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. Rough numbers per year obtained by simply looking at the target complex would be three each to Mars and Venus, two each to Jupiter, Saturn, and the asteroids, and one each for solar probe, out-of-ecliptic, cometary, Mercury, Uranus, and Neptune or Pluto missions. The other six probes would serve engineering missions to supply data needed for the design of the manned ships to follow.

Scientific personnel should not be bothered with this engineering-data requirement; but it is important if we are not to have another data deficiency such as occurred when Apollo had to be designed with practically no lunar-surface information. Then, it is not inconceivable to me that the Mars-probe program could be run over by a Mars 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. 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 exceeding 7 yr even to the deep planets. It would also open the launch windows to Mars and Venus to almost half a year. The use of close flybys of Jupiter would permit 90-deg out-of-ecliptic and close solar-probe missions, and would reduce the maximum travel time to about 5 yr. The Russians used a close flyby of the Moon for Lunik III trajectory control in 1958.

Provision of 18,000-fps velocity capability in the spacecraft coupled with clever guidance and atmospheric braking 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 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. This amount of improvement can be achieved by high-energy chemical propellants. A 450,000-lb three-stage hydrogen–fluorine launch vehicle with 3000-lb spacecraft could deliver 500 lb of payload in all missions.

For the whole solar system, it should be possible to carry over 150 lb of scientific instrumentation besides structure, controls, communication equipment, and power supply within the 500-lb payload. 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 would represent a high enough rate to effect production-line savings in vehicle, spacecraft, and launch costs.

The system should be useful for at least 10 yr, since only the advent of large-scale manned exploration or gaseous-fission-powered spaceships to project scientific payloads could economically replace it.

A strong effort for such a modern, cost-effective vehicle should be mounted. If only a 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 of about 50% of the current budgetary outlay for space science.

The alleged requirements 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. Any future program should not be pursued merely as a Mars and Venus exploration, but reoriented along the lines of this discussion. Otherwise, it will prove another limited-objective development.

There is a fertile field for cooperation with the Soviet Union in unmanned scientific exploration of the solar system. A basis for planning cooperation earlier than commonly thought feasible exists in the system advanced here.

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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