SOLAR TRANSPORTATION

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Introduction

Let us leapfrog to the fall of the year 2000, or 35 fiscal years, from FY 1966 to FY 2001. By doing so, we will be able to describe the status of solar transportation in our time as well as to look back at the events of the past three and one-half decades which produced the advanced state of interplanetary travel which we enjoy at the turn of the millenium. We have made much progress in exploring the solar system and understanding its history as well as the present condition of its bodies. It has been the age of discoveries all over again, and, if we include the exploration of the Moon, it has been a repetition of those first thirty and some years which characterized the period at the beginning of the first age of discovery.

In fifteenth century Portugal, Prince Henry the navigator, sponsored the art of seafaring and navigation. This resulted in the discovery of the Cape of Good Hope in 1488. Christopher Columbus discovered the Bahama Islands on his first voyage in 1492. In 1497 Vasco Da Gama reached India; and in 1500 Cabral, the coast of Brazil. Ponce De Leon visited the Florida peninsula in 1513. The Magellan expedition finally circumnavigated the globe in 1519 to 1521, 33 years after the discovery of the Cape of Good Hope. Yet, in spite of what had been accomplished, countless discoveries remained to be made through the centuries to follow. Utilization of the discoveries made, had barely begun in the world of 1521. In our world of the year 2000, countless discoveries still remain to be made in the solar system and we too have hardly begun to utilize the worlds we have incorporated into the realm of human activities.

Solar Transportation in the FY 2001

Today, in the fall of 2000, the interplanetary flight corridors from Mercury to Saturn are alive with manned vehicles of relatively luxurious and sophisticated design, driven by quite advanced propulsion systems. Unmanned probes have approached the Sun as close as 0.15 AU (Figure 1). Large and very advanced unmanned probes have reached out as far as the planet Pluto (Figure 2), and at this moment rove the vast, dark regions

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of trans-Pluto and interstellar space. All of this far flung traffic and the conditions of the vehicles, manned or unmanned, are continuously monitored not only by greatly expanded (compared to FY 1966) deep space networks on planet Earth, but also, since FY 1988, by a large Lunar deep space control facility (Figure 3). An array of relay satellites in near-Earth and cislunar space has been established, converting practically the entire region between Earth and Moon into one gigantic antenna system designed to monitor traffic in and even beyond the solar system. Automatic monitors have been placed into suitable orbits in the inner and outer solar system.

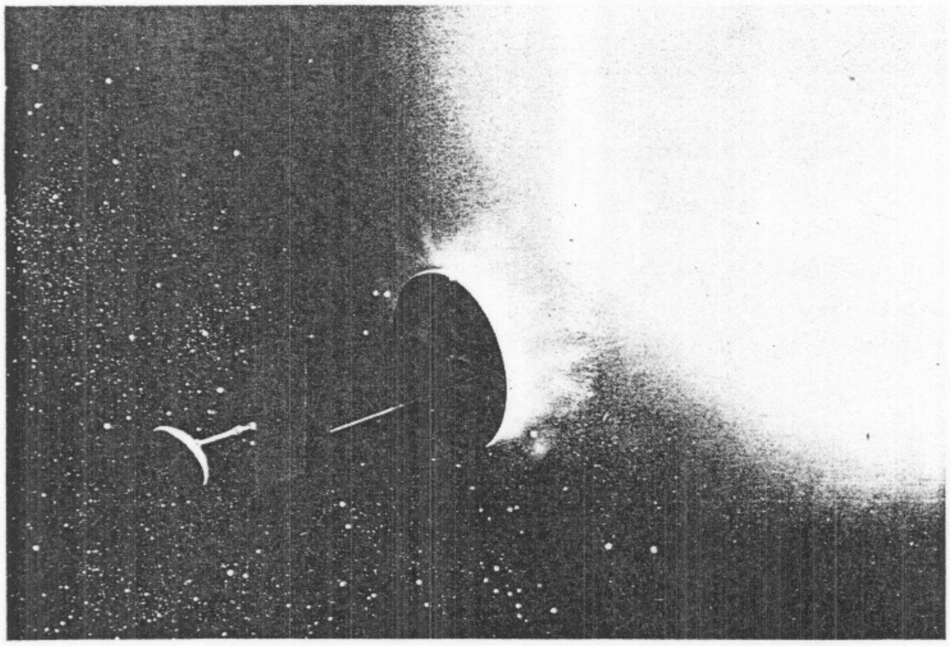

Figure 1. Advanced Solar Probe Designed for Very Close Approach to the Sun. Probe is Precisely Attitude Controlled to Point Evaporation Heat Shield Toward Sun, Shielding Equipment With the Shadow. Probe Uses Thermoelectric Power Generation, Radiation Cooling Surfaces in Shadow of Heat Shield and is Equipped with Plasma, Particle and X-ray Sensors.

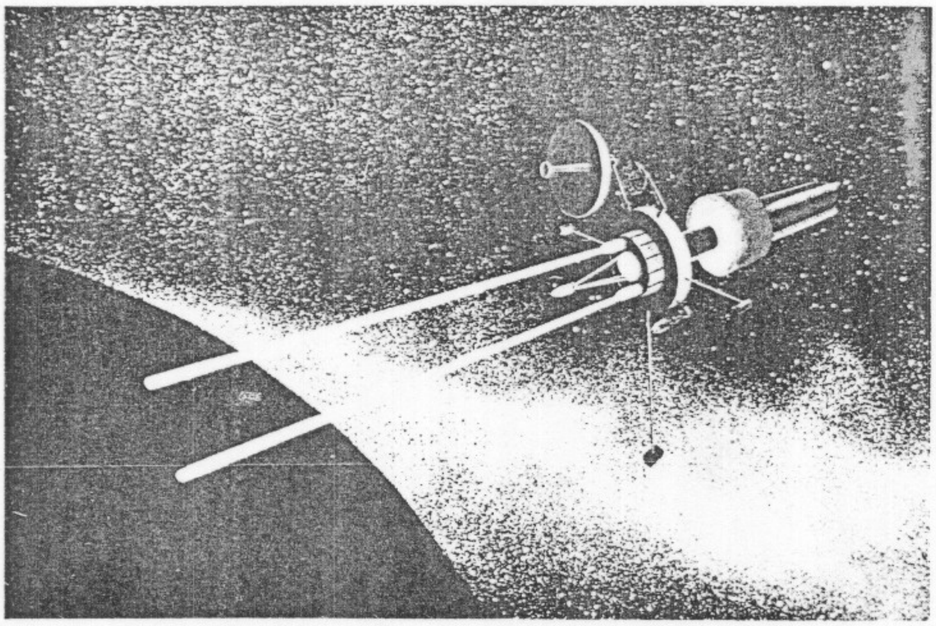

Figure 2. Unmanned Probe Approaching Pluto. Probe is Powered by Thermionic Radioisotope Power Generator, Two Laser Beams for Surface Illumination, With Optical Sensors Slaved to the Beams. Other Equipment Comprises Cosmic Radiation Counters as well as Field, Plasma and Particle Sensors.

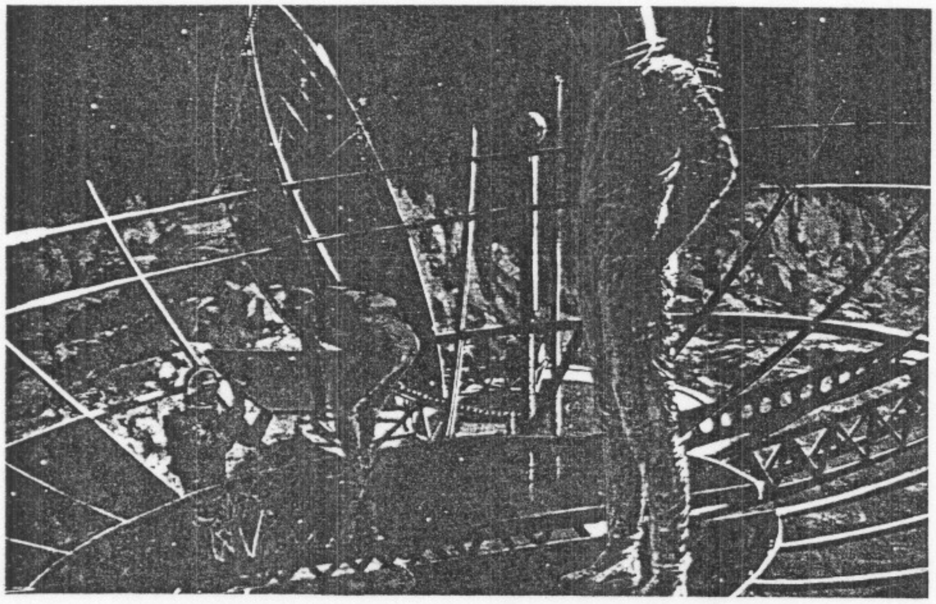

Figure 3. Establishment of Lunar Deep Space Communication Network and Radio Telescope in 1985 Through 1988.

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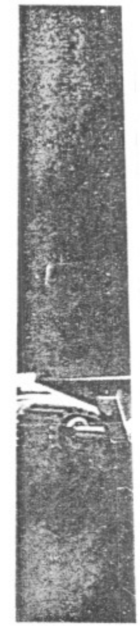
We have rendezvoused with, and planted an automatic scientific station on, the asteroid Icarus, which approaches the Sun as close as 0.169 AU, or about 47 percent of the distance of Mercury, and which swings out beyond the orbit of Mars to an aphelion distance of 1.68 AU. Our helionauts, as these men who fly our large interplanetary vehicles call themselves in this era of continuing specialization, have covered the solar system from the sun-scorched shores of Mercury to the icy cliffs of the Saturn moon Titan. They have crossed, and some have died doing so, the vast asteroid belt between Mars and Jupiter and have passed through the heads of comets. Owing to the pioneer spirit, the courage and the knowledge of our helionauts and of those engineers, scientists and technicians behind them, astrophysicists today work in a solar physics station on Mercury (Figure 4); biologists experiment on Mars (Figure 5), backed by a well supplied research and supply station on the Mars moon Phobos; planetologists have landed on Venus; and teams of scientists right now study what has turned out to be the two most fascinating planets of our solar system, Jupiter and Saturn, from research stations on Callisto and Titan (Figure 6).

As you know, we also have begun to utilize some of the discoveries. Our metal ore mining and processing facilities on Mercury are just three years old. On Mars, a long range program has just been started to induce in the circumpolar regions of the northern and southern hemisphere, large scale cultures of special Mars-hardened plants, the result of twenty years of biological and agricultural research on Earth, on the Moon and on Mars proper. These plants are suitable for human consumption. While initially they will support the growing research base on Mars, it is expected that, within the next 50 years, Mars will export foodstuffs to Earth.

The traffic between Earth and Mercury, Earth and Mars, and Earth to Jupiter has become heavy enough to warrant the establishment of an orbital supply and rescue station at Venus. This station has worked successfully and has saved lives during the past eight years. Venus is a particularly useful place for a helionautical "coast guard" station, because this planet's orbital elements complement those of Earth for missions to Mercury as well as to Mars, Jupiter and beyond. For example, the synodic period of Earth relative to Mars has an average value of 780 days, meaning that a given transfer corridor from Earth to Mars or Mars to Earth occurs roughly every 780 days. The average synodic period of Venus relative to Mars is 337 days. A transfer corridor to Jupiter from Earth is available roughly every 1.1 years. Transfers between Venus and Jupiter, or vice versa, are feasible every 234 days. Mercury can be reached from Venus more rapidly, in cases of emergencies, though not more frequently than from Earth, because of differences in angular velocities. The gravitational field of Venus can frequently be utilized to reduce transfer energy requirements both ways between Mercury and Earth, and at less frequent occasions, between Earth and Mars; and occasionally, also on Earth-bound transfers from Jupiter in conjunction with a powered maneuver.

Our Earth orbital and Lunar facilities now handle some 24 interplanetary vehicles annually; that is, arriving and departing (Figure 7). This may not sound like much compared to our hypersonic and sub-orbital build-up in air traffic, as well as to our orbital and Lunar traffic. It does mean, however, that every month two large interplanetary vehicles are being processed; and the number is rapidly rising. While this could not have occurred without a great variety of technological advancements in practically every area of space technology, the present traffic is due primarily to the advancement of the nuclear pulse drive, or NP, and the controlled thermonuclear reactor drive, or CTR. The energy provided by these two propulsion systems is at last commensurate with the enormous performance requirements of interplanetary travel. Thereby, mass ratios are kept in bounds; interplanetary shuttle services can be established, the interplanetary vehicles revised for many missions and the Earth-to-orbit logistic demands kept within reasonable limits. While these propulsion systems did not pioneer interplanetary flight, they nevertheless made sustained solar transportation economically possible; and they are the basis for the present high and rapidly climbing level of helionautic traffic (Figure 8), much as the jet engine was the basis for the upswing of air traffic in the 1950's and 1960's of the past century.



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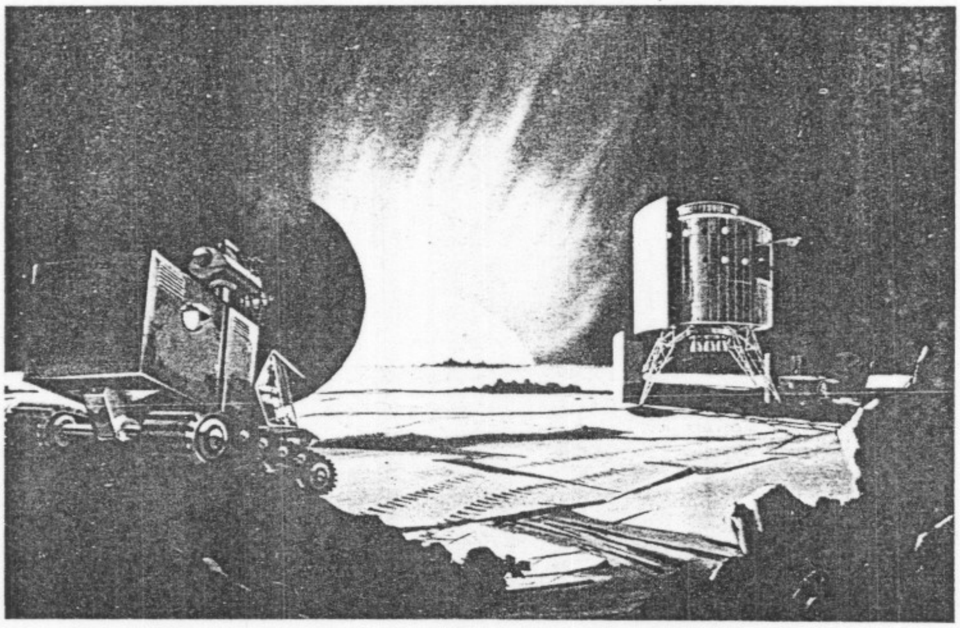

Figure 4. Solar Physics Research Station at the North Pole of Mercury, 1988. Both the Landing Module and the Crawler are Protected by Highly Reflective Radiation Shields.

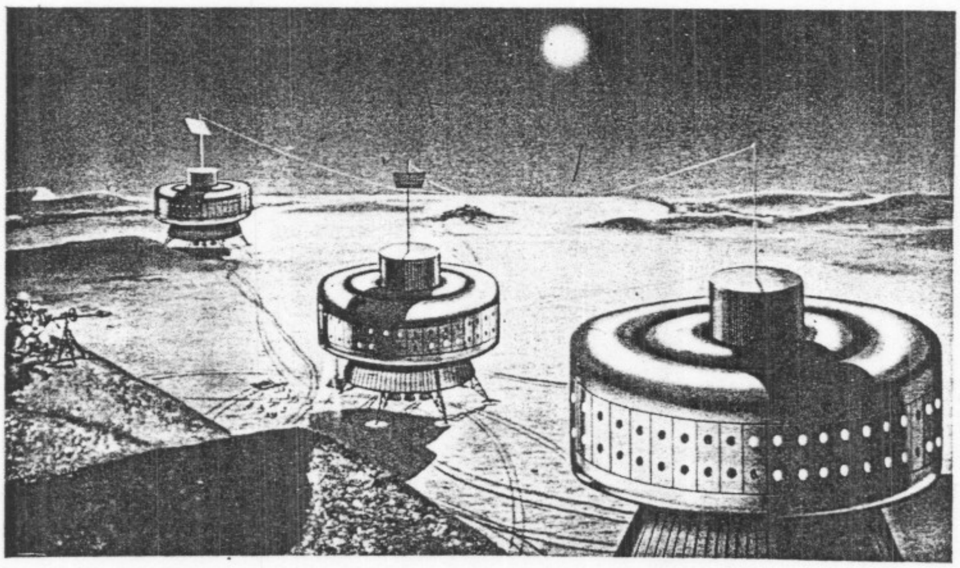

Figure 5. Astrobiological Research Base on Mars, 1992. The Three Base Modules Shown are Supplied with Electrical Power from a Nuclear Power Generation Module Seen Partly Buried in the Background. Earth is Visible to the Upper Left of Sun.

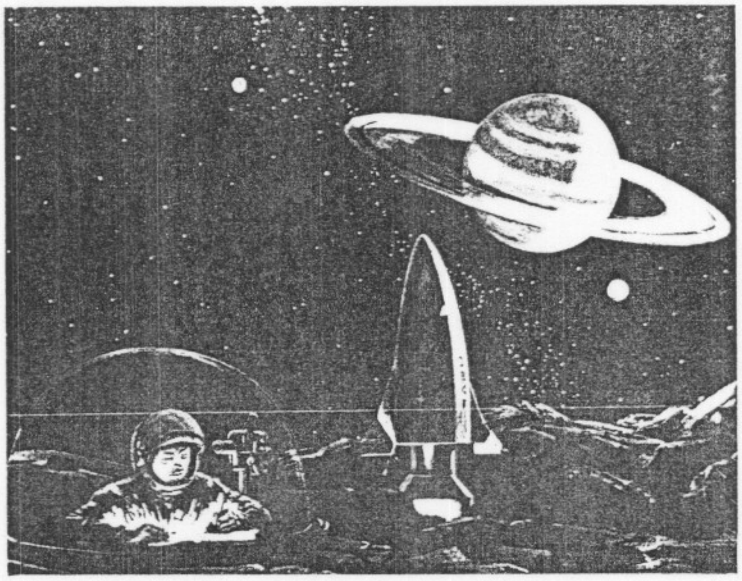

Figure 6. Establishment of Saturn Research Station on Titan in 1995.

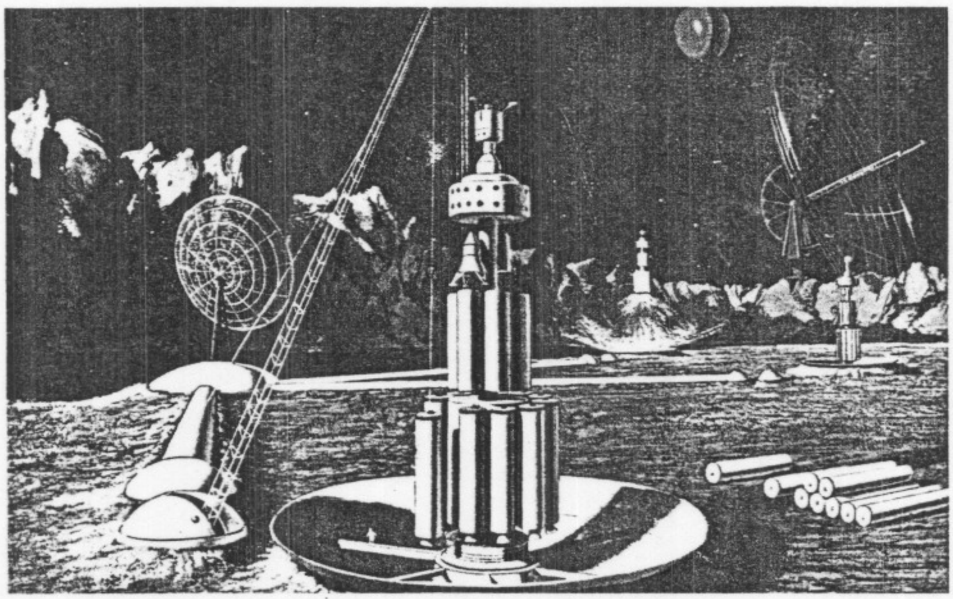

Figure 7. Earth-Moon Based Planetary Space Port in 1988. Spacecraft are Nuclear Pulse Propelled Interplanetary Vehicles, Launched by Solid Propellant Lift-off Rockets Side-Mounted Around the Lower Cylindrical Section Which, Like the Cylinders at the Spacecraft's Center Section, Contains Nuclear Pulse Units. In Background a Large Antenna, Belonging to the Lunar Deep Space Network is Visible.

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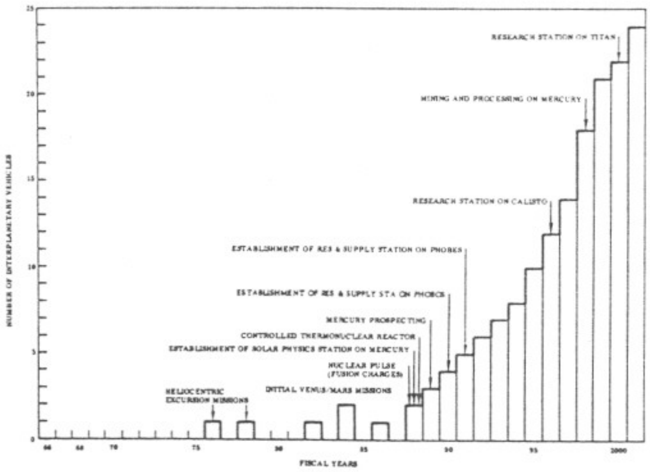

Figure 8. Number of Interplanetary Vehicles Handled Annually by Earth-Orbital and Lunar Facilities (Arrival and Departure).

The present volume of solar transportation, however, is not a result of technological progress and associated missions alone, impressive as they may be. It reflects, rather, the status of the visions of nations and, indeed, of most of mankind. Therefore, before we review in retrospect the progress in space technology and space flight, let us take a brief look at the world of the fiscal year 2001.

Our World of the Year 2000

Behind us lies a century the like of which can have occurred only a very few times, namely, at the most crucial turning points, in the history of mankind. This century saw the final phase of a long process in which man's agricultural, feudalistic and static social order was transformed into an industrial, techno-scientific, socially fluid and dynamic system. In this phase, changes followed each other with unprecedented speed and scope; deep rooted conflicts which gradually accumulated in the wake of the preceding three centuries of transformation were resolved with explosive violence, cruelty and hatred.

We, in the year 2000, look back at the twentieth century as the years in which the new era was finally born after centuries of incubation in the minds and hearts of great men and women of many nations. The twentieth century is the gulf which separates the last century of the old era and the first century of the new one in which values, outlooks and frames of reference are quite different. The hour of birth, be it of a life or of an era, is the hour of truth in which pain, doubt and fear challenge, and the intensity of their onslaught causes the compensating forces of strength, confidence and bravery to rise to rare peaks of intensity and power. The world seems to break apart under the agony of this unmerciful confrontation of the old and the new. The great symbols of the space age, namely, rocket technology, nuclear technology and modern electronic technology were borne in the dark days of World War II. But, since war can never bear peace, the rockets remained missiles, the nuclear devices remained bombs and the radar did not cease to be the ear which was anxiously listening for the signal of death from the hostile world of "the other side". The past was lost, the future not yet won; and mankind shivered in the feverish chill of hostility, hatred and death-fear unleashed in the succession of wars and confrontations.

These were the realities.

Throughout those years, a small group of people of many nationalities, while facing those realities, refused to surrender their vision of missiles-turned-spacecraft, of nuclear power becoming a means of propelling space vehicles to other worlds and of radar waves reporting exciting discoveries from deep space. What they suggested seemed at first impractical, inconsequential and without utility or payoff. But we now know that they had built their case on the solid foundations of long-range logic and realism. In a way, they invested in certain blue chip stocks of human nature, namely in man's desire to build not only a safer and more convenient world, but also a greater and ever expanding world with unlimited challenges and opportunities to apply his newly acquired powers. The investment payoff was bigger than even they might have expected. Space became a very real challenge to man; and there was no way back to the old days. There never is.

At the beginning, the space age was primarily an extension of the cold war rivalry between the United States and the Soviet Union. But space is much bigger than man; and as he moved ever farther out, the race concept became, shall we say, a bit inadequate. Vast, positive goals have a way of replacing hostility by cooperation. By exploring the great universe around him, man can indeed be delivered from the terrible paralysis of hatred and fear.

The therapeutic value of positive goals is known to every psychologist. Improving education, conquering diseases, providing better living conditions are positive goals. But space exploration is a stronger goal, simply because it is one of the reasons which makes these other goals worthwhile. For, what good is it to create better men and women without providing an opportunity for them to apply their better qualities? Space exploration is one of those opportunities. Indeed, the wars and confrontations of the twentieth century have shown what the advancement of science, technology, medicine, general education, more housing and better roads can do if there is not, at the same time, a generation of greater goals toward which the new mankind can exert itself. In the bygone era, the great goals were the conquest of hunger, disease, illiteracy, slavery and poor living conditions in general. These are no longer primary goals, but "mop-up operations". Space is one of the great primary tasks of the new era. Space operations, be they applied to exploration or applications, have become a symbol of that certain attitude of friendly rationality which alone can bear peace, if people and nations apply it in dealing with one another.

The challenge of distance and of worlds beyond his own has always exerted a magic influence on man, causing him to overcome even the most powerful fears, born out of the superstitions of his time and to plunge into the unknown. No matter how this drive is rationalized by establishing a causality with certain apparent utilities at that time, there remains an important basic influence which is emotional and which is rooted in the deep-seated obsession to penetrate the mysteries of nature and to absorb them into a system of human understanding. This unquenchable thirst for knowledge and understanding is perhaps the third of man's basic drives. While he shares two others - hunger and sex - with all life on Earth, the third is his alone and sets him apart from the other creatures as being endowed with a mind which must forever feed on the unknown or die. The unknown is the preferred challenge after all, if compared to the business of coercing and killing his own kind. Crossing established frontiers of the known world, mentally or physically, is mankind's way of maturing and is one of the few fundamental causes of terrible crises and of true and lasting happiness known. Man entered space as an Earth-oriented being. Now, space begins to transform him into a cosmically oriented being with a broader and more mature outlook at his own small planet and the problems of living on it.

Why am I dwelling so long on these fundamental aspects of space flight? Do not all of us, in the year 2000 consider them self-evident? Oh, but it is important that we remind ourselves occasionally of the fact that they were not self-evident forty or fifty years ago; that they were verified and proven to us in many ways between then and now; and therefore, in the clarity of hindsight became self-evident to us.

Remember the friendships between American and Russian astronauts in the early 60's? Remember when we started training astronauts from major European nations and Japan; and the Soviet Union did the equivalent on their side in the 70's; and when, I believe it was in 1976, a party consisting of a Russian, a Polish and a Czech cosmonaut on the Moon risked their lives to rescue a US - British - Italian team that had crash landed with their Lunar Hopper some distance away from the Soviet outpost? Or, when in 1978 a US - German - Japanese crew in one of those old-fashioned, but then brand-new oxygen-hydrogen cislunar space vehicles abandoned their mission and, in a brilliantly executed series of maneuvers saved the life of a Russian-Rumanian cosmonaut team stranded in Lunar satellite orbit? Can we, another and more fortunate generation, truly appreciate the impact which these brave acts had on the minds of people all over the world?

We learned not only to rescue and help one another. "Proliferation" of astronautics was welcomed as much as proliferation of nuclear weapons was feared. Satellites of many nations began to form a virtual super-structure above the surface of our planet; forestry, plant disease detection, weather observation, communication, air and sea traffic control have become matters of global concern and, gently, almost imperceptively, encouraged and strengthened cooperation between nations.

These were the formative years and they brought about much enlightened international legislation regarding the use of and mutual assistance in space and on the Moon.

Yet the nuclear bombs and the manufacturing of nuclear weapons continued by the members of the "nuclear club". The trend I have described really moved into the big time with advent of economical interplanetary flight which, as you know, requires nuclear propulsion. Opening the solar system finally furnished an "energy sink" big enough for the excess energies and prestige desires of any and all nations. You will remember that China became a major member of the space-going fraternity in the second half of the 70's. As an ambitious world power, China finally began to appreciate the importance to its prestige, of channeling some energy into spaceflight efforts. By now its efforts are comparable to those of the USA and the Soviet Union. In order to understand that, we must recall the changes in population size, agricultural and industrial developments which took place in the last forty years.

In the years between 1960 and 2000, the population of the U.S.-led Western group, including Japan, grew from roughly 0.9 billion to 1.4 billion. The Russian-led group grew from 0.4 to 1 billion and China from 0.7 to 1.6 billion. The population in the developing countries rose from 1 billion to about 2.4 billion.

The "heart lands" of the before mentioned groups are the USA, the USSR and China. The areable area of the world has a bit more than doubled by the year 2000. In addition, massive use of insecticides, large scale replacement of animals by machines, improved fertilization and improved irrigation has increased the yield and added food for 1 billion more people. The utilization of the oceans has been greatly improved. Aside from the many direct improvements, additional growth has been achieved by the use of satellites in determining and controlling sea food production. On land, satellites have also contributed to improved food production and reduced spoilage by weather control, forest fire control, early detection of plant diseases and other means of this type. Space ecological technology which made rapid progress in developing food production for the astronauts in orbit and on the Moon, has greatly improved the potential of producing food in non-areable surface areas of the Earth. Therefore, feeding 6 to 7 billion people in the year 2000 is not an unsolvable problem.

Looking at the combination of agriculture and industry in the year 2000, we find that in the USA, 4 million people in agriculture produce food for about 320 million Americans; that is, a ratio of 1 to 80. In the USSR, the ratio is approximately 1 to 25, i.e., 20 million feed 500 million Russians. In China the ratio is about 1 to 15 - that means that in China, 100 million people produce food for 1,500 million Chinese. In spite of the still considerable differences in efficiency, these numbers mean that the U.S. has available an industrial army of approximately 120 million people, the USSR 180 million people, and China has an industrial army of 600 million. Thus, China can be industrially less efficient than the U.S. and still constitute an economic power at least equal to the U.S. This economical power, together with political ambitions, desire for prestige, practical necessities connected with being a leading world power; and the fact that the economy of Russia and China still is not as consumer-goods oriented as that of the western world, are the reasons why not only the U.S. and with it in partnership western Europe, and the Soviet Union, but also China are vigorously participating in the exploration of the solar system and the beginning exploitation of its resources.

The need for nuclear propelled interplanetary space vehicles led to a revision of the nuclear test ban treaty, and the development of nuclear pulse. (NP) propulsion. Freedom to develop nuclear pulse units led to rapid obsolescence of the original NP versions, which used fission energy, and their replacement by systems using more efficient and cleaner fusion energy. Today, by far the majority of the bomb-manufacturing capacity of the United States, Russia and China, is used to produce charges for NP drives of interplanetary vehicles. I find it impossible, indeed, to come up with a more real and more practical benefit of the space age so far.

Military satellites and manned space laboratories occupied by military personnel were developed and also are in use today. But they have never assumed the threatening posture which is so characteristic of ground armament. Indeed, in that form, weapons are an anachronism in space, for practical as well as for spiritual reasons, though technologically and physically space certainly offers a formidable potential of warfare against Earth. Over the decades, military space installations have become, or remained, systems with control, surveillance, inspection and verification functions and no one on Earth now considers them a threat. This, in turn, caused certain restrictions to control weapon developments to be relaxed gradually in cases where they began to interfere with non-military space developments and programs, such as in the case of the nuclear pulse development. The resolution of the UN General Assembly in October 1963, adopted unanimously, expressed intention of all parties to refrain from stationing weapons of mass destruction in space. In the middle 70's, China joined this resolution. While far less formal than the many, monumental disarmament conferences of the twentieth century, this resolution probably is the only one on record which was never violated.

The Origin of Solar Transportation in Retrospect

During the early 60's a new approach to decision-making in the field of astronautics was introduced, namely, the establishment of a national goal in space. This prevented the national space capability from going after too many lesser goals at a time, thereby either requiring an unrealistically large budget or splitting its resources so many ways that none or only few of the goals could be attained and that overlap and redundancy of effort seriously threatened the effectiveness with which these resources were deployed. The decision to declare manned landing on our Moon by 1968 a national goal, led to the first major national space effort, the Apollo Program, which subsequently formed the basis for far more projects than just landing two men on the Moon. The Apollo technology accelerated the advent of a comprehensive Earth-orbital program; it formed the basis for continued Lunar exploration, from orbit as well as on the surface, to the point where an intelligent decision regarding the size and type of a permanent scientific Lunar base could be made. Apollo made available Saturn V to the unmanned planetary exploration program; a launch vehicle which was powerful enough to absorb readily the weight increase of the Voyager probe when Mariner IV measurements showed the Martian atmosphere to be more tenuous than assumed originally; and which was capable of injecting deep space probes of 14,000 lbs to heliocentric parabolic escape velocity. Last, but not least, Apollo and its follow-on activity, the Apollo Applications Program, laid the foundation for a manned interplanetary program.

In 1965 members of the National Academy of Sciences conducted studies to advise the government on scientific aspects of the national space program. So far as planetary and interplanetary exploration was concerned, their recommendations were; (1) to start in the 1965-1975 period a shift of emphasis toward the planets and away from the Moon, progressing toward a roughly equal expenditure for Lunar and planetary exploration in the 1975-1985 period; (2) to give primary emphasis to Mars, secondary emphasis to Venus and the major planets. Their anticipation was that unmanned experiments would probably provide the most significant contribution to the program of planetary exploration in the 1965-1985 period.

In 1963 to 1967, NASA, supported by industry, had carried out extensive planning research covering the entire spectrum of post-Apollo activities, including the prospects of manned interplanetary missions. NASA knew that the heritage of the Apollo Program, a vast concentration of talent, huge ground facilities, supporting industries and flight hardware production capabilities, would have to be passed on to an adequate follow-on program, if the nation's leading space posture was to be maintained. On the other hand, it was apparent that a new major national goal had to be recommended to the President and to Congress, subject to certain constraints. Two important constraints were the budget limitations and the need to avoid selecting a program for whose execution proper foundations had not yet been laid. Manned missions to other planets belonged in this latter category. The planning of the manned orbital and Lunar programs for much of the 70's required two important decisions. First, there was the question whether the operational plans for the late 70's and the 80's should include manned interplanetary missions. This question was answered positively. The second decision pertained to the question of whether the first mission should be planned for the late 70's or the early 80's. This question was more difficult to answer. A decision was dependent upon the type of mission and the willingness to meet the development requirements needed to bridge the gap between existing and required state-of-the-art.

It was recognized that a helionautical mission capability imposes considerable requirements on the space transportation system and on the destination payload which could not be met immediately at the state-of-the-art level of 1970.

Among several deficiencies, the two major constraints of the 70's were no doubt the limitation of specific impulse of interplanetary vehicle propulsion to the 400-800 sec range; and limited as well as expensive Earth-to-orbit logistic capability, compared to the true needs of an interplanetary era. Due to these two constraints, helionautical transportation system requirements were highly mission profile dependent. This fact, in turn, motivated considerable research in the flight mechanics of helionautical missions.

The Evolution of Helionautic Mission Profiles

In helionautics, a wide trade-off range exists between (round-trip) mission period and (overall) mission velocity, defined as the sum of the velocity changes associated with all principal maneuvers (i.e. not including navigational corrections and capture orbit trimming), including gravitational losses. A heliocentric round-trip mission profile consists of two or more transfer profiles, defined as a section of a heliocentric orbit connecting two points in the Sun's gravitational field by means of a Keplerian orbit section. A group of transfer profiles (between the same two points) feasible during a specific and limited time period (e.g. one month) is referred to as a transfer corridor, specified by transfer time, departure window and arrival window at the target. In a given transfer corridor, transfer time and velocity requirements are related such that a reduction in transfer time usually causes an increase in transfer velocity. Originally, only slow transfer orbits (transfer angle around 180 deg) were considered (1925-1, 1953-1). Figure 9, which illustrates such a mission profile to Venus and to Mars, shows that very long capture periods, referred to as

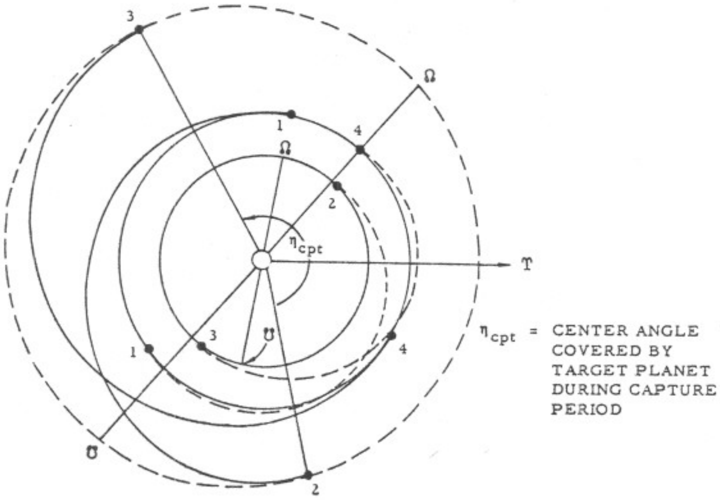

Figure 9. Long-Transfer Synodic Mission Profiles to Venus and Mars

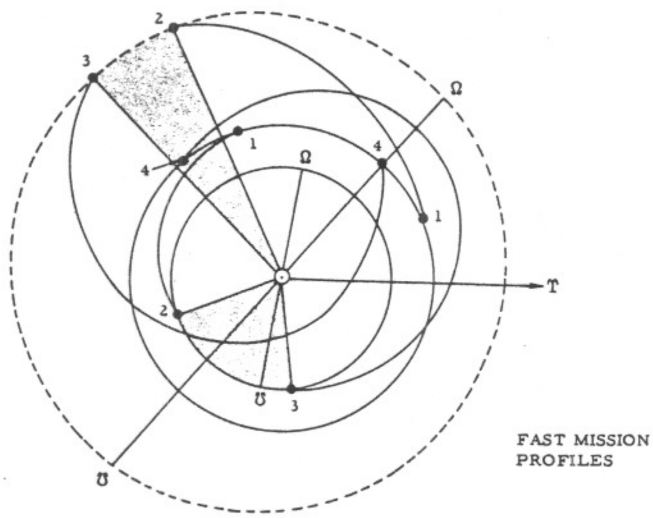

Figure 10. Fast Mission Profiles to Venus and Mars

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synodic capture periods, are involved (periods between points 2 and 3 in Venus and Mars orbit, respectively). In 1957 the investigation of fast round-trip missions began (1957-1, 1958, 1959). It was recognized that the mission time should be reduced primarily by reducing the long capture period (and only secondarily by reducing the transfer time); and that capture periods could be reduced only by making the outbound transfer orbit different from the Earth-bound transfer orbit. This method of variation of transfer orbits led to mission profiles which were characterized by a short outbound and a long (not slow) Earth-bound transfer path (Figure 10). For Mars in particular, these mission profiles have the disadvantage of resulting in very high Earth return velocities (50,000-75,000 ft/sec, depending on mission year). The Earth-bound transfer path leads to perihelion distances between 0.65 and 0.45 AU, for the lower and higher Earth return velocities. This is shown in Figure 11 which shows the heliocentric distance of the inner solar system planets versus time and flight profiles, corresponding to the approximate center line of the transfer corridors for fast and slow missions and, in a few cases, for missions of intermediate duration (550-700 days). Favorable transfer corridors for fast round-trip missions to Venus or to Mars are presented in Figure 12. In view of the very large Earth approach velocities upon return from Mars missions, it appeared originally that only two alternatives would be available: Either a retro-maneuver at Earth approach; or very high speed atmospheric entry. The first alternative tended to increase the Earth orbital departure weight (ODW) of the space vehicle to a degree which was unacceptable in view of the available logistic capability from Earth. The second alternative required acceptance of a fairly hazardous terminal operation, great strain on the crew after having been under low or zero-g conditions for the preceding 400-450 days and the expensive development of a very high speed Earth entry module (EEM).

In 1963 it was suggested that the retro-maneuver can be applied far more economically near the perihelion (perihelion brake, PB) (1963-2). The resulting transfer path change is shown in Figure 13. For celestial mechanical reasons, the Earth approach velocity v_∞ is reduced more effectively at the perihelion than at Earth approach. This led to the study of

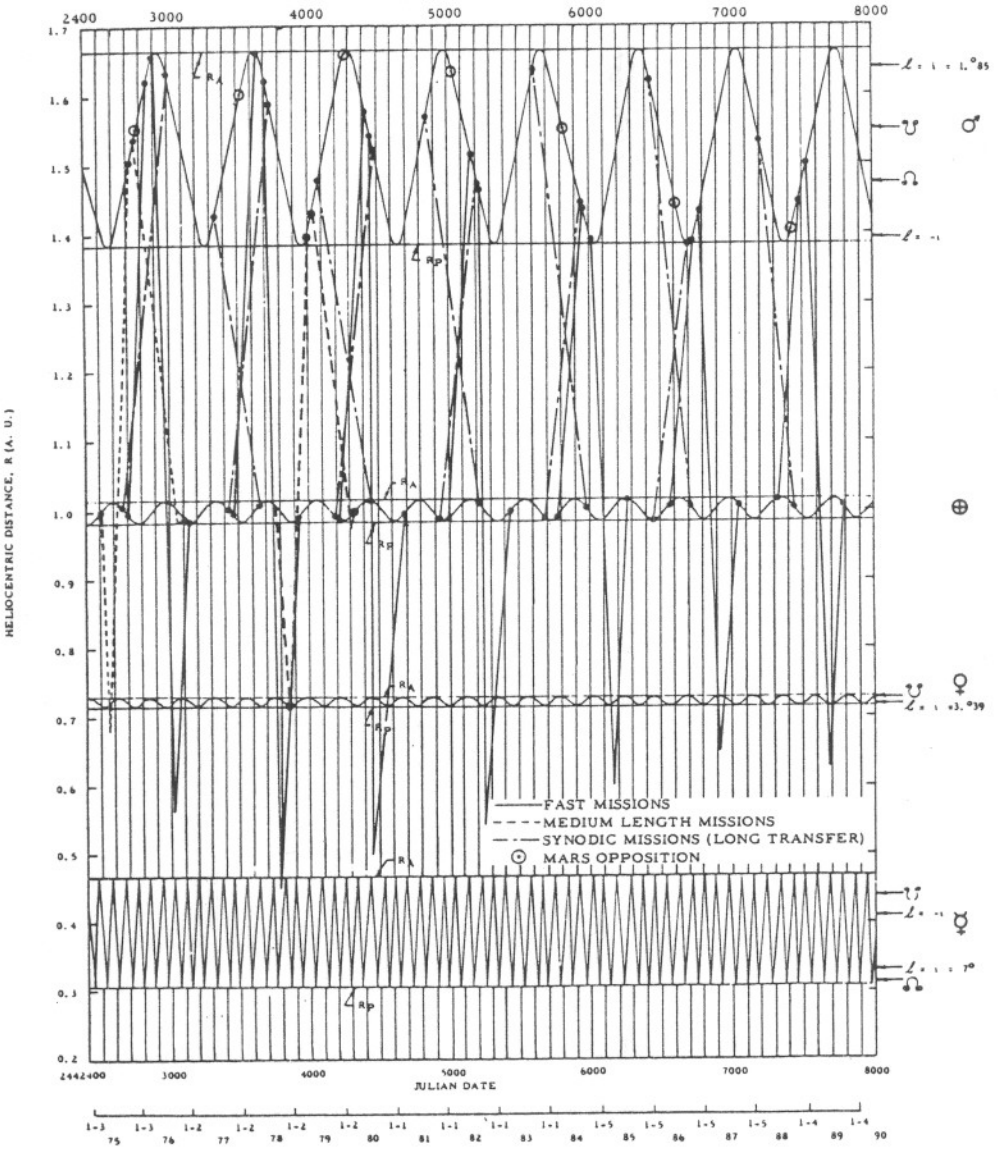


Figure 11. Lines Representing the Center of Transfer Corridors for Fast Missions and Long-Transfer Synodic Missions to Mars and Back

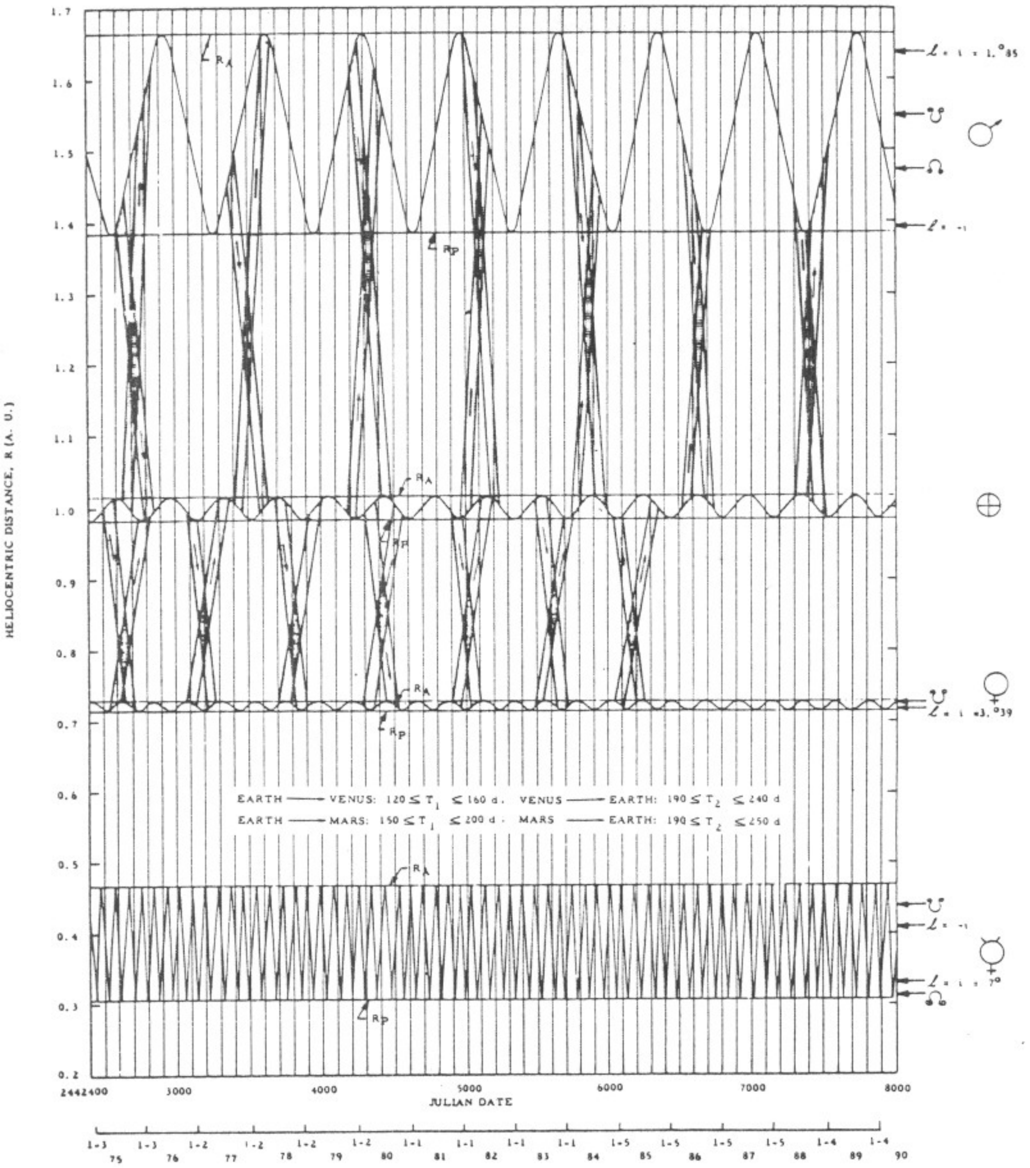


Figure 12. Survey of Favorable Transfer Corridors for Mono-Elliptic Transfers Between Earth and Venus and Between Earth and Mars

still other maneuvers in heliocentric space, but the PB maneuver remains till today the most effective one in this family (1965-2). Application of the PB maneuver reduces the Earth entry velocity v_E at return from Mars from the values shown by the white columns in Figure 14 to those shown by the superimposed black columns at a PB maneuver V_{PB} which is seen to be distinctly smaller than the Earth approach maneuver v_E required to achieve the same reduction in entry velocity. If, prior to the PB maneuver, the interplanetary vehicle was relieved of all payload mass no longer needed for the remaining 60 to 90 days of flight, a significant reduction in ODW could be obtained, especially if a low-weight solar heat exchanger drive of 700 to 800 sec specific impulse was used.

But still another alternative was found which consisted of using the gravitation field of planet Venus. In 1954 the use of the gravitational fields for changing the flight path of cislunar or interplanetary vehicles was suggested (1954-1). In 1956 a double fly-by mission, involving Mars and Venus was described (1956-1). In 1963 a number of bi-planet capture missions were investigated (capture at both planets) (Figure 15) involving Mars-Venus or Venus-Mars missions and a Venus-Mercury mission (1963-3), based on the philosophy that, if a favorable transfer window does not exist between planets A and B, it may exist between A and C, and subsequently between C and B. Figure 16 shows transfer corridors, both ways, between Venus and Mars (1964-4). By combining the transfer corridors shown in Figure 12 with those in Figure 16, transfer corridors for bi-planet missions are obtained, involving Venus and Mars (Figure 17). These mission profiles show that sometimes the best planet sequence is Venus first Mars second, sometimes the reverse sequence. While these bi-planet capture missions would not necessarily reduce the mission velocity, compared to a one-planet capture mission, they would not cause an increase either,[1)] thereby permitting the visit of two planets essentially for the price of one. But an increase in mission period from 100-200 days had to be accepted. In 1963/64 the final step was taken[2)] in the development of the bi-planet mission analysis, namely, to combine the perturbation maneuver concept and the bi-planet capture mission concept (1963-4, 1964-1,-2,-3, 1965-1,-3, 1966-1). By reducing the capture period at Venus to zero, coming from Mars and going to

1) Based on Earth entry velocity range and return from Venus.

2) Independently by several researchers.

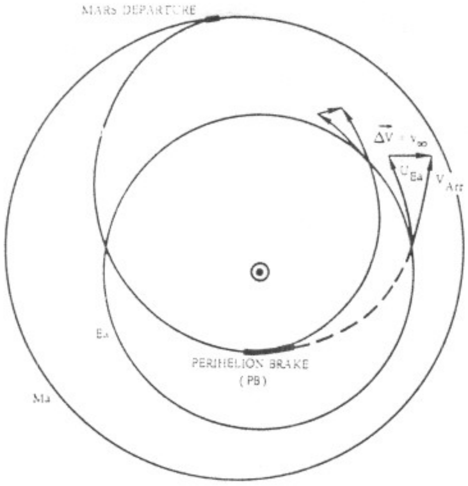


Figure 13. Perihelion Brake Maneuver and Its Effect on Earth Arrival Velocity

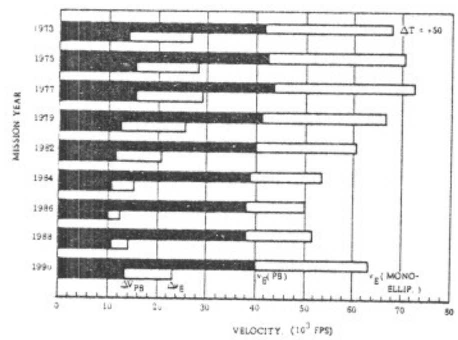


Figure 14. Reduction of Atmospheric Entry Velocity at Earth from Value Following Mono-Elliptic Return, V_E(Mono-Elliptic) to V_E(PB). Comparison of Velocity Changes Required if Reduction is Achieved by Means of Perihelion Brake (Δ_{PB}) or by Means of a Geocentric Retro-Maneuver (ΔV_E).

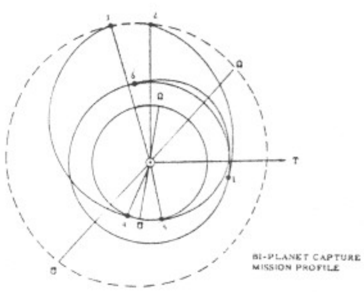


Figure 15. Bi-Planet Capture Mission Profiles to Venus and Mars

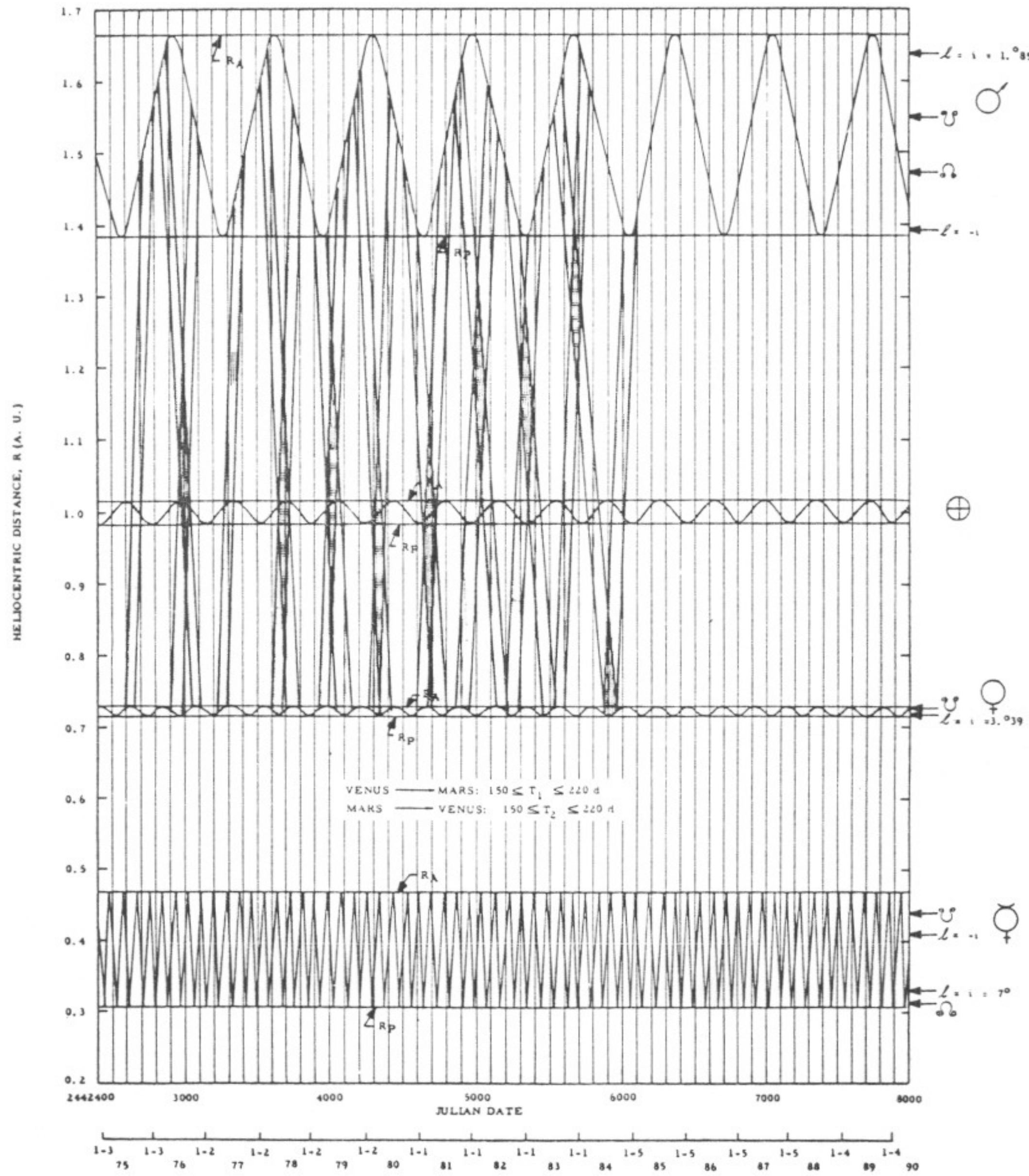


Figure 16. Survey of Favorable Transfer Corridors for Mono-Elliptic Transfers Between Venus and Mars

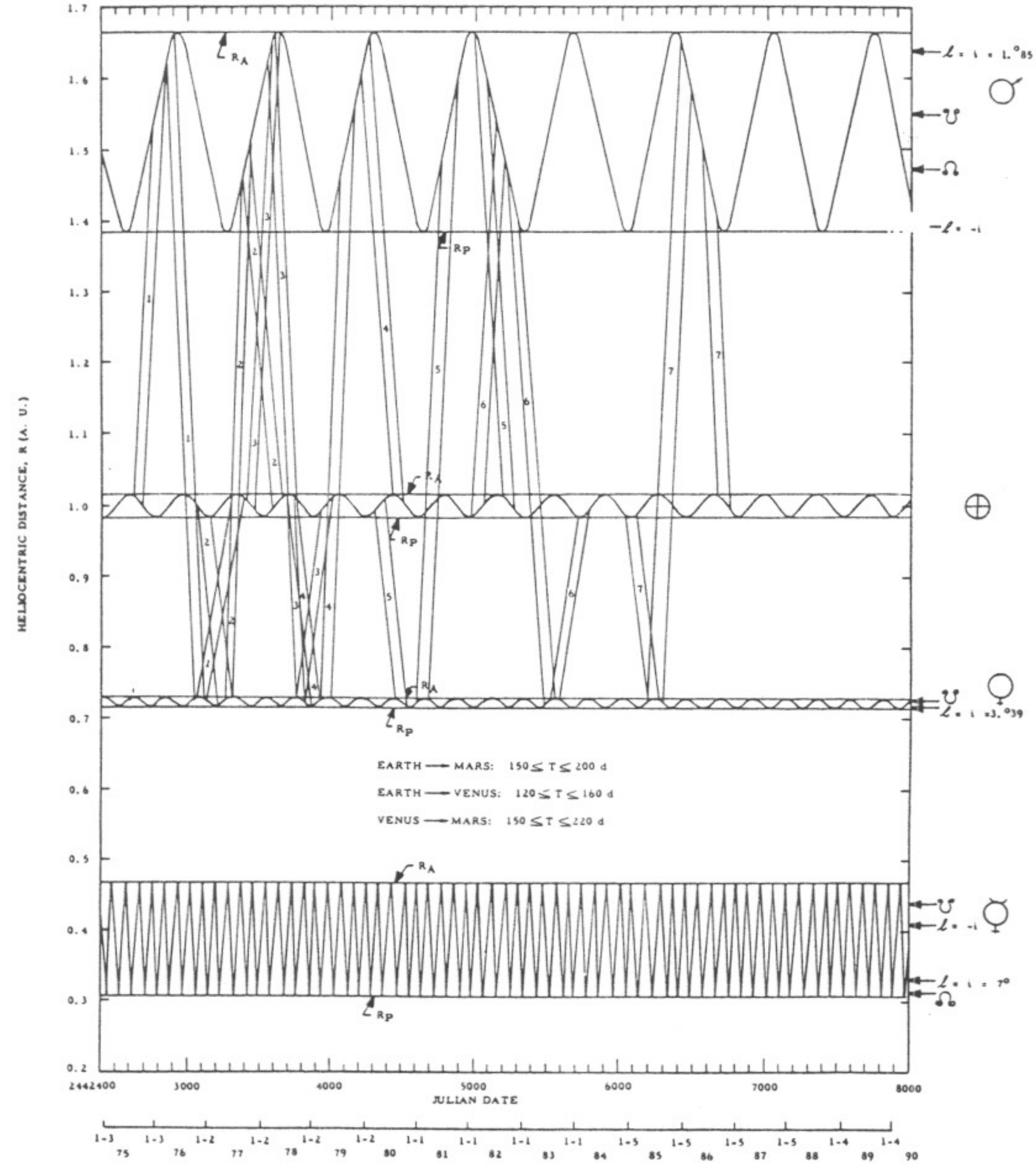


Figure 17. Favorable Transfer Corridors for Bi-Planet Venus-Mars Missions Involving Capture or Powered Fly-by Near Either or Both Planets

Earth, one has a Venus fly-by mode on return from Mars which would, under certain conditions, reduce the Earth approach velocity to some 40,000 to 43,000 ft/sec; in other words, attain with little or no propellant expenditure what in the PB maneuver after all required some propellant mass. The price for the Venus fly-by (swing-by) mode resulting in such low Earth arrival velocities is a longer mission period (550-600 days instead of 400-450 days). A Mars capture, Venus fly-by mission profile is shown in Figure 18. Typical mission data are presented in Figure 19 (1966-1). For comparison, Figure 20a shows typical data for mono-elliptic round-trip missions (direct flight) to Mars and back associated with the mission profile shown in Figure 10. Venus, Earth and Mars form the same constellation every 6.4 years (1962-1, p. 1069; 1966-1). Due to this commensurability, each round-trip mission pattern recurs approximately every 6.4 years.

In summary, mono-elliptic fast round-trip missions to Mars (420 to 460 days) require a mission velocity of 43,000 to 53,000 ft/sec if the mission is terminated by unretarded atmospheric entry at Earth. For retardation to 50,000 ft/sec entry velocity, the mission velocity varies between 45,000 and 70,000 ft/sec. For retardation to about 40,000 ft/sec terminal entry velocity and perihelion brake during return flight, the mission velocity varies between 50,000 and 68,000 ft/sec. For Mars mission profiles involving Venus fly-by and terminal entry velocity of about 40,000 ft/sec, the mission velocity varies between 36,000 and 50,000 ft/sec. For Venus round-trip missions of 380 to 440 days duration with 10 to 40 days capture period, the overall mission velocity varies between 45,000 and 50,000 ft/sec for circular capture orbit at $r^* = 1.1$ planet radii and between 30,000 and 40,000 ft/sec for capture in an elliptic orbit with periapsis $r_p^* = 1.1$ and apoapsis at $r_A^* = 8.8$ ($n = r_A/r_p = 8$). These data are surveyed in Figure 20b and a representative range of mission velocity requirements is indicated which had to be met by the early interplanetary vehicles if a capture mission terminated by Earth atmospheric braking was to be performed. At 60,000 ft/sec mission velocity, the majority of the mission opportunities in the 70's and 80's could be utilized. At a mission velocity of 40,000 ft/sec or less, the number of available opportunities would be reduced sharply, unless the mission period would be increased to values between 600 and 1000 days.

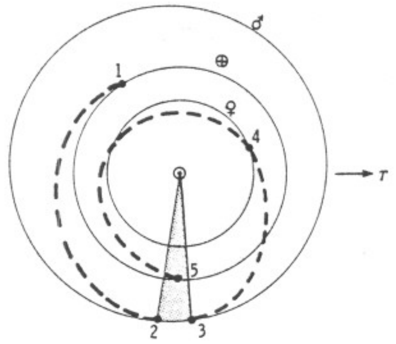
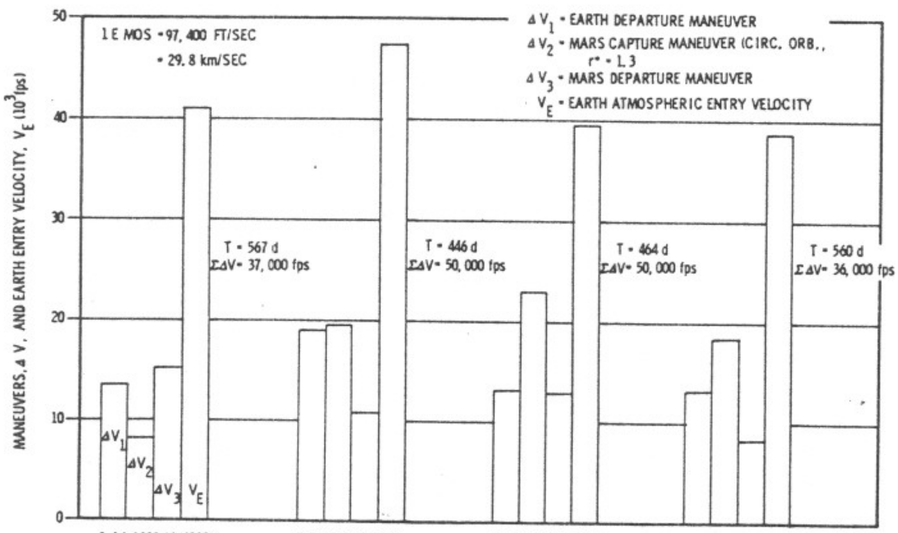


Figure 18. Mission Profile Involving Mars Capture
and Venus Fly-by on Earth-Bound Transfer



DEP EARTH	1-14-1982 (4-4983)	7-9-1983 (4-5524)	2-16-1984 (4-5746)	3-21-1985 (4-6145)
TRANSF. TIME (d)				
TO VENUS		54		171
TO MARS	229	202	156	184
CAPT. PERIOD MARS (d)	28	20	18	29
TRANSF. TIME (d)				
TO VENUS	149		162	
TO EARTH	161	188	128	176
Ar Ea	8-4-1983	6-9-1984	5-25-1985	10-2-1986

Figure 19. Velocity Profiles for Bi-Planet Missions Involving
Capture at Mars and Fly-by at Venus Either Outbound
or Earth-Bound

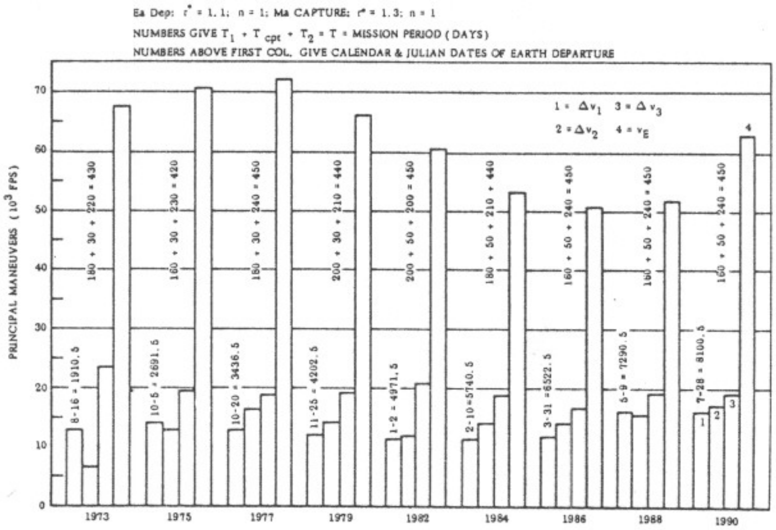


Figure 20a Velocity Profile for Mono-Elliptic Mars Capture Missions 1973-1990. In Each Mission the First Three Columns Represent the Maneuver Required at Departure from Near-Earth Orbit, Mars Capture in Circular Orbit at 1.3 Radii Distance and Mars Departure Maneuver, Respectively. The Fourth Column Shows Atmospheric Velocity at Earth Approach Along Hyperbolic Path Without Braking Maneuver.

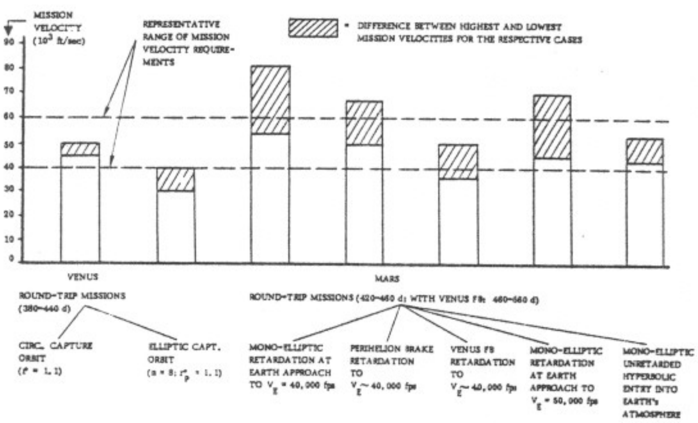


Figure 20b Comparison of Mission Velocity Requirements for Initial Round-Trip Missions to Venus and Mars With Capture at Target Planet and Termination by Entry into Earth's Atmosphere.

Some Early Problems, Alternatives and Decisions

So far as the transportation systems were concerned, the following major requirements had to be taken into account:

- An appropriate Earth Launch Vehicle (ELV)
- Space Taxis (ST) and associated orbital auxiliary vehicles
- Long duration ecological system (semi-closed)
- A set of mission modules referred to summarily as Life Support Section (LSS), including radiation shelter, command module with a diagnostic and maintenance system, and ecology module (containing the before mentioned ecological system) and other modules, such as repair ("workshop") module and a module for data handling, data transmission and electric power generation (EPG).
- Orbit Launch Preparation Modules (OLPM), supporting the mating, fueling and checkout activities which, during the orbital preparation period, are directed from the LSS.
- An Earth Entry Module (EEM) for velocities up to at least 42,000 ft/sec, if not 50,000 ft/sec
- Propulsion Modules for the interplanetary vehicle (heliocentric interorbital space vehicle, HISV)
- An Orbital Tanker Vehicle (OTV), to supply the HISV propulsion modules with the fuel which could not be carried during their transport into orbit, and with made-up fuel owing to unavoidable losses in orbit.
- A Destination Space Vehicle (DSV) if secondary (excursion) missions are planned at the destination.

Interplanetary vehicles, whose mission duration requires from 400 to 6000 days, must have an extensive on-board checkout and repair facility, located in the Life Support Section (LSS). By placing the LSS into orbit at the beginning of the orbital assembly process, this section serves as the orbit launch facility. In its initial form the LSS has two modifications, compared to its mission configuration which is shown in Figure 21a.
Orbit launch preparation modules (OLPM) are attached; and an LSS maneuvering propulsion module occupies the space in which the mission version carries the Earth Entry Module (EEM). The Earth assembly configurations of the LSS configuration are shown in Figure 21b.

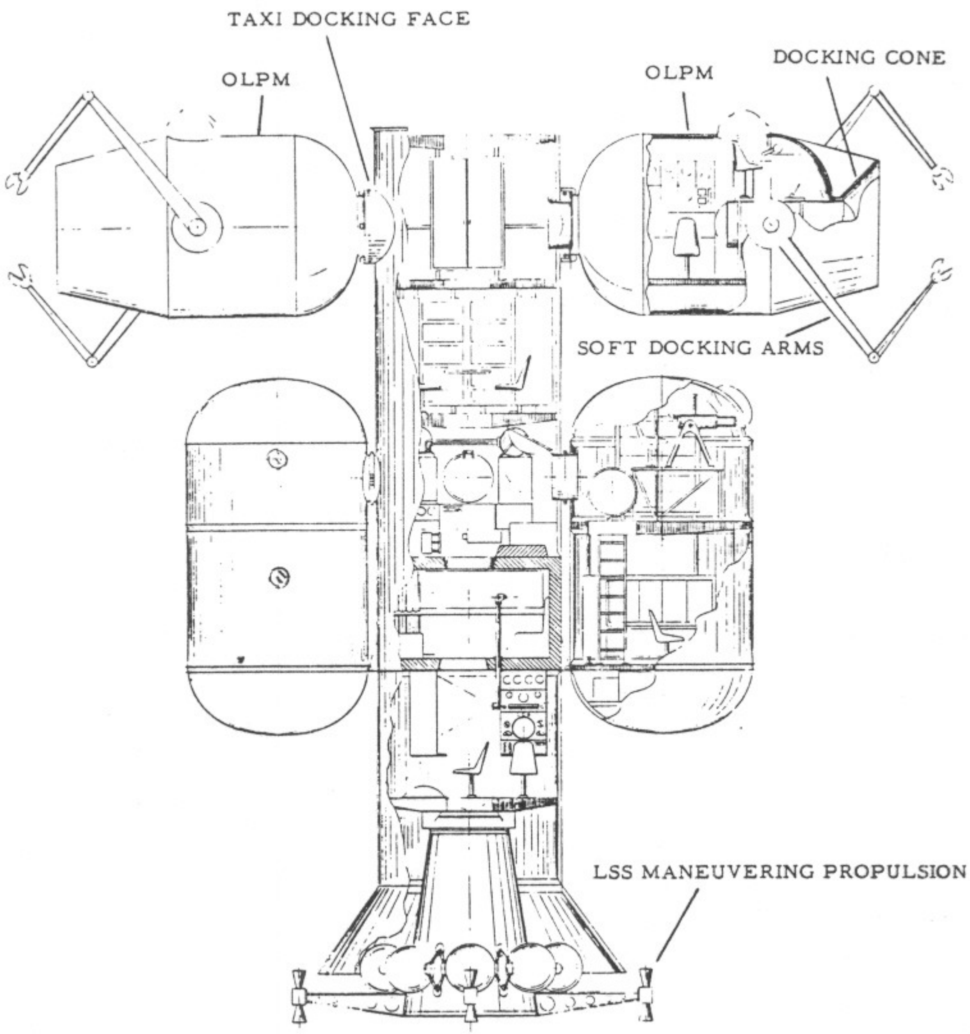


Figure 21a Radial Life Support Section: Earth Assembly Configuration, Acting as Orbit Launch Facility. Orbit Launch Preparation Modules (OLPM) will for Mission, be Replaced by Taxis. LSS Maneuvering Propulsion Module will be Replaced by Earth Entry Module (EEM) (Reference 1964-12).

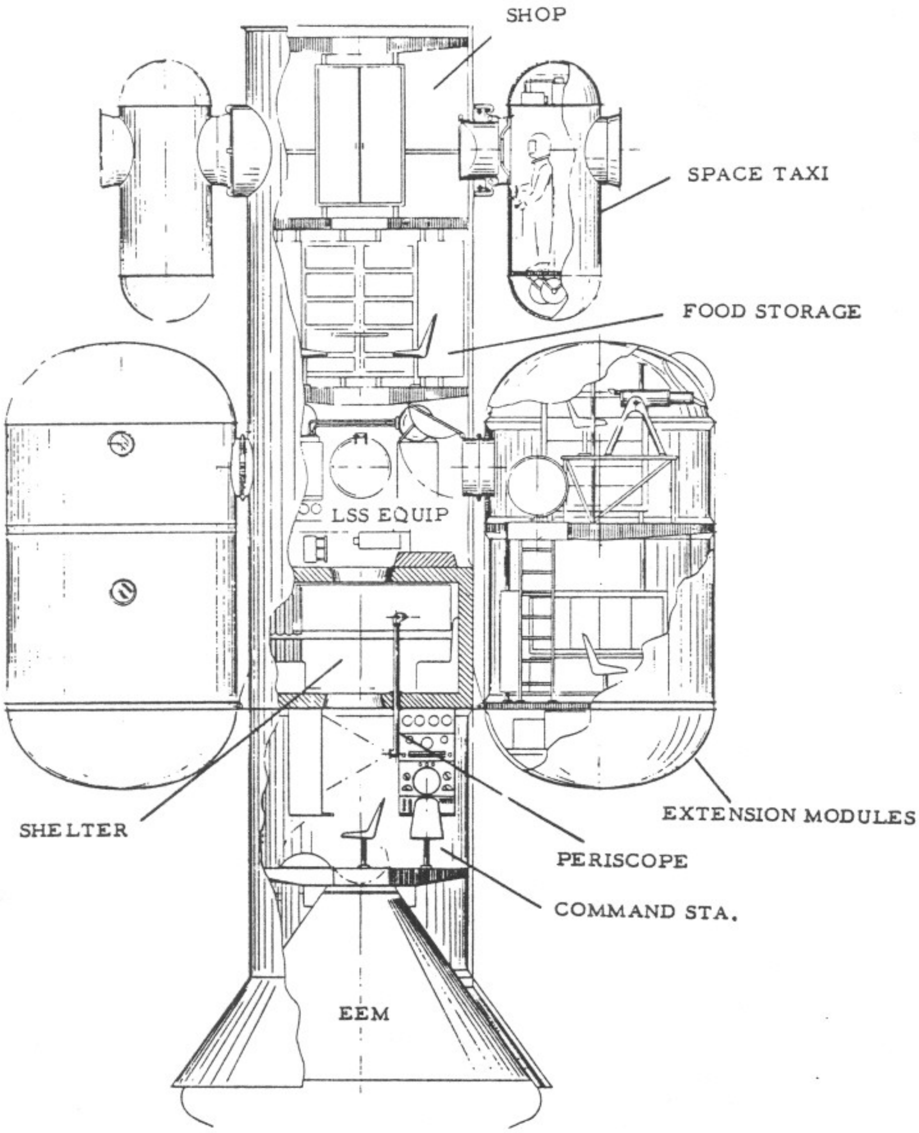


Figure 21b Radial Life Support Section: Mission Configuration. Earth Entry Module (EEM) is Located at Forward End of Interplanetary Space Vehicle. Interface with Propulsion Section is Rearward of the "Shop". Extension Modules are Jettisonable. (Reference 1964-12).

The broad mission objectives of planetary exploration can be divided into three groups: (1) Reconnaissance and surveillance from orbit; (2) Deployment of unmanned probes from the manned space vehicle in orbit; (3) Surface excursion. The first and second objectives can be met by a planet fly-by or by a capture mission profile. The third objective requires a capture mission.

So far as destination payload requirements were concerned, the case was simplest for Objective (1) and a capture mission, because planetary surveillance and reconnaissance technology in the late 60's was already highly developed for planet Earth. A more complex situation and lower data yield was encountered in a fly-by mission. The situation was similar in the case of Objective (2). Here, too, the capture mission was most compatible with experiences previously gathered by unmanned surface probes from Lunar orbit. The surface excursion naturally was the most demanding mission.

Some of the key destination payload requirements are indicated by the following list, covering fly-by to advanced landing missions.

- Optical and radar reconnaissance equipment for the planet fly-by or capture mode.
- Data selection, compression and storage for surface excursion, and/or capture missions where the data influx (especially the optical) can be extremely large (10^{12} to 10^{14} bits in a period of 20 to 30 days). Storage requirements are obviously far less for the fly-by mode.
- Protection of crew members from extraterrestrial life (surface excursion missions). If a planet is approached which might harbor life, a capture mission preceding a surface excursion mission is likely to be mandatory. The capture mission with its possibilities and time to survey, to land unmanned probes practically anywhere on the planet and to exploit their findings in great detail from relatively close range, may make it possible to arrive, with a high degree of confidence, at conclusions as to existence, nature and anthropological or terrestrial compatibility of alien life, without exposing crew members to unknown conditions. Therefore, this portion of the destination payload includes auxiliary probes

such as floaters (if feasible), landers (stationary or mobile) and returners, capable of returning to the orbiting manned vehicle, carrying samples of soil and specimens. In 1968/70 it could not be predicted whether life existed on Mars; and if so, whether the limited number of landings of Voyager Landers on specific points of the Mars globe would be able to ferret it out. The planning of manned Mars missions, therefore, had to be prepared for the case that the question of micro-life on Mars was still not yet resolved at the time of the first manned flight to that planet.

- Shelters
- Base modules

From these two lists it is quite apparent that the difference between the (then) existing and the required state-of-the-art would be effected decisively by preceding activities in the manned orbital and Lunar programs. Important general contributions are in these respects:

- High launch vehicle delivery reliability
- High reliability of orbital mating and fueling
- Availability of trained orbital launch assist crews
- Advanced long-duration ecological systems
- Improved quality of space structures (minimum of gas leakage, heat leakage, advanced thermal control, etc)

Benefits from the development of cislunar transportation systems included:

- High reliability of the interorbital space vehicle
- Advanced flight control and other components of the operational payload equipment
- Microminiaturization of electronic equipment
- Minimization of electric power requirement
- Long-term storage of cryogenic fluids in chemical and nuclear propulsion in space

Benefits which resulted from the development of a Lunar surface operations capability included:

- Advanced long-duration ecological systems
- Well developed and tried shelters and base modules
- Radioisotope electric power generation systems

Additional contributions, especially in the areas of radioisotope power generation systems, electronic microminiaturization, interplanetary navigation and flight control, minimization of electric power consumption and knowledge of the environmental conditions to be expected in the target planet's environment, its atmosphere and, where feasible, its surface, became available from the unmanned probe program.

The technology and capability provided by the Apollo program in the area of launch vehicle, orbit launched oxygen-hydrogen stage, orbital test facilities; and satisfactory progress in the development of the first nuclear engine (NERVA) brought, by the end of the 60's, the following questions to the point of a decision:

- Should a manned planetary mission be based as closely as possible on the Apollo technology? If so, the mission would have to be a Venus or perhaps a Mars fly-by mission.
- Is a fly-by mission worth the effort in view of the fact that a two-Voyager-in-one-launch capability on Saturn V was being developed (first launch 1973 window to Mars); and would the effort to be expended for such a project not delay the advent of a far more worthwhile (in terms of mission yield over cost) capture mission, using (at least for Earth departure) the nuclear engine which would be available well in time for capture mission? Fly-by mission periods are quite comparable with capture mission periods. The crew size for both missions is comparable and the amount and type of destination payload is comparable in type, though perhaps not in quantity. The major difference between the two mission modes lies in the overall velocity requirement, hence in the orbital departure weight of the interplanetary vehicle; and therefore in the logistic requirements (and cost) for the launch preparations in orbit (mating and fueling of modules). On the basis of comparative studies, it was concluded that at least 75 percent of the development effort for a capture mission (without manned surface excursion) would have to be expended also for a fly-by mission; but that only about 50 percent of the launch preparations effort would be required. Earliest time period for the fly-by mission would be the 1977 to 1979 time period, a period of high to maximum

solar activity. Waiting till early 1982 would bring into play not only nuclear powered engines and less severe mission conditions, but also a larger and more economical post-Saturn launch vehicle and more data from Voyager and unmanned interplanetary probes.

A manned fly-by mission would have two primary and one secondary objectives, so far as the non-technological mission objectives are concerned. The first of the two primary objectives would be to carry the unmanned probes to the target planet before they are launched, rather than have them launched from Earth directly. During the transfer, the transfer and the probes, particularly their payloads, are being maintained and possibly even prepared (e.g. being sterilized) by the mission crew. This "delivery mode" is likely to add to their mission success. The second primary objective would be to utilize the presence of the crew and the possibility of observing the target planet prior to arrival for the purpose of obtaining more selective surface delivery; that is, more precise aiming of Landers to specific places on the planet where the information obtained would be more valuable (e.g. scientifically useful) than if the Lander would descend on another point. This presumes, of course, that enough is known about the planet to begin with to make a distinction between "more" and "less" worthwhile landing sites. For Venus, and possibly even for Mars, such detailed knowledge might not yet be available by the time a fly-by mission is launched. In order to gain such knowledge, the mission objectives of unmanned Venus and/or Mars probes in the years preceding the first manned fly-by mission would have to be focused primarily on Orbiters which provide detailed information on a global scale, rather than on Landers; except where Landers would be needed to provide engineering data for the design of the Landers carried aboard. These Landers, however, could be lighter, simpler and less costly than the Landers which would be developed for scientific purposes in connection with the Voyager Program. It can be seen, therefore, that a decision to carry out a manned fly-by mission in the 70's could effect profoundly the planning and mission objectives of the Mars and Venus probes of the Voyager class. The secondary objective would be for the mission crew to directly acquire as much information as possible during the fly-by process. This objective is designated as secondary here not because it is less important, but because the comparatively brief period of fly-by permits only a very limited extent of exploration by the mission crew directly; and, therefore, the main data acquisition in this type of manned mission would still have to be provided by unmanned probes. The beneficial effects as to their mission objectives would, therefore, constitute logically the primary objectives of the manned mission. Clearly, the question as to whether to organize a fly-by mission in the 70's rather than concentrating on a more comprehensive capture mission first thing in the 80's would be subject to complex decision criteria, because from the technological viewpoint alone, either alternative appeared to be feasible from the standpoint of the planner in the late 60's. The question also emphasized the need, pointed out years earlier by advanced planners, for integrated planning of the unmanned and manned planetary mission programs.

- Should a decision to launch the first manned planetary mission in 1982 eliminate any heliocentric mission activity which would keep the public stimulated and interested in the manned planetary program during the 70's while being at the same time technologically and operationally meaningful? This question was decided negatively, in that it was determined that at least one heliocentric excursion mission should be planned for the 70's. In this mission the spacecraft is injected into an escape path from Earth, leading into an elliptic flight path relative to the Sun. This ellipse may contain that transfer path section which later would actually be flown on the way to the target planet. In any case, the spacecraft would fly only a relatively small section of the heliocentric ellipse with an associated flight time of between 10 and 50 days, depending upon the performance capability of the vehicle. At the end of the outbound time period, a heliocentric maneuver would be carried out which causes the spacecraft to break away from the outbound flight path and follow a new elliptic course, leading to interception of, and return to Earth. This mission simulates a heliocentric abort operation as it would, under actual emergency conditions, have to be executed by a Venus- or Mars-bound vehicle which is forced by some trouble to abandon its mission at a point of the transfer orbit at which it is not yet committed to the mission, i.e., still possesses an Earth return capability. Depending on the type and performance of the vehicle, this point may be anywhere between 10 and 70 days following Earth Departure.

These conclusions, though not attained unanimously, matched those of the scientific community regarding the importance of a precursory unmanned probe program prior to deployment of manned vehicles; and to the extent that engineering and mission worthwhileness considerations by and large tended to discourage the undertaking of a manned planetary mission in the 70's. However, the conclusions differed from the scientific recommendations insofar as they recommended a manned mission early in the 80's (1982). This would require a heavier expenditure for the development of such a manned mission capability in the 70's than would otherwise be needed. The logic of these findings was supported by the fact that, if Mars was to be the primary target, one should be prepared to utilize the 1984 and 1986 windows (which offered very favorable mission conditions, at a time of the quiet Sun) to carry out a landing mission, should the results of the Mars Voyager experiments and of the 1982 manned capture mission justify such an undertaking.

The National Goals 1970-1985

In early 1969, the principal national goals were announced for the time period of approximately 1970-1985. They were

- Exploration of the Solar System with unmanned probes and attainment of a manned mission capability to Venus and Mars.
- Establishment of a permanent scientific laboratory on the Moon.
- Establishment of functional (or applications) space stations in near-Earth orbits and in 24-hour orbits.

Specifically, the steps toward achieving these goals were formulated somewhat along the following lines:

(1) Manned orbital installations, as represented by such concepts as resupplied small orbital research laboratories (SORL) and the associated orbital operations complex (OOC), which has the function to develop an orbital docking, mating, inspection, fueling, checkout repair and deep space launch capability, are the key to post-Apolo extended manned lunar and planetary capabilities. Achievement of these capabilities should have first priority within the manned space flight program during the first half of the 70's

(2) The SORL is also the bridge to larger permanent functional space stations which utilize the orbital operations capability and the near-Earth space and low-g environment for the benefit of man and for scientific purposes. Such installations, summarily referred to as large orbital research laboratories (LORL), are the logical goal of the 1975 to 1985 period.

(3) In the manned Lunar program, a permanent scientific laboratory by 1982 to 1985 is the primary objective. For this purpose, extended Lunar surface excursions and preparatory developments for the scientific station are to be carried out throughout the 70's. This development of a Lunar surface capability is also of greatest consequence to achieving the capability of executing possibly a Mars landing mission by the middle 80's.

(4) In the field of unmanned planetary probes, Voyagers to Mars and Venus are given the highest priority, for scientific reasons as well as their importance in obtaining environmental data for engineering purposes, and sufficient scientific data to plan worthwhile experiments, for the first manned mission. The principal launch vehicle for these planetary missions is Saturn V.

(5) Second in importance is the preliminary exploration of Jupiter and Saturn by a precursory probe, referred to as Advanced Planetary Probe (APP), in the 70's, utilizing the opportunity of reaching Saturn via Jupiter fly-by in the second half of the 70's.

(6) In the area of manned planetary flight, the capability of orbiting Mars in the early 80's and, depending on the scientific worthwhileness, landing on Mars by the middle 80's is the principal national objective. Precursory missions into cislunar space and heliocentric excursion missions, as well as an appropriate development program of solar probes and solar monitors constitute the supporting mission operations in the 70's. Principal propulsion systems for the manned interplanetary vehicle are the nuclear engine NERVA II, based on hydrogen heating by a solid core nuclear reactor, chemical modules using oxygen and hydrogen and, for maneuvers in heliocentric space, a solar heat exchanger drive, heating hydrogen by means of solar energy. Principal launch vehicle is post-Saturn.

(7) In view of points (1) through (3) and (6) the Earth-to-orbit personnel traffic and the load delivery requirements will experience a steady and significant rise throughout the 70's and 80's. Consequently, reuseable launch vehicles, more than any other space transportation

vehicle, become the key to the economic execution of program goals (1) through (3) and (6). Two developments are needed, namely, a reuseable orbital transport (ROT) for personnel traffic and a reuseable, flexible-payload post-Saturn of three to five times the payload capability of Saturn V. Both vehicles are chemically powered, nuclear propulsion being restricted to manned, orbit launched cislunar and heliocentric interorbital space vehicles. Development goals are an operational ROT by 1976 and post-Saturn by 1980.

This famous 7-point program of the American President outlined the next major round of the space program of the United States and its European and Japanese partners. It contained the provisions for the origin of solar transportation.

The Selection of Propulsion Systems for Extended Manned Flight Through the Solar System

It was not an accident, but programmatic necessity, that the second round (1970-1985 period) contained a greater variety of national goals than the first round (1962-1970). Specifically, three reasons account for it: (1) The need for orbital assembly operations for planetary missions; (2) The 50 to 70 times longer mission period to planetary targets far less well known in detail than the Moon; (3) The need to maintain the Lunar program of exploration and base establishment, and to utilize the new environment for scientific, technological and other applications purposes.

When the time came for defining the third round (1985 to 2000), helionautics had come of age and the new national goals could be defined without close involvement with the orbital and Lunar program. The definitions for this period had to be made by 1980/81.

With flights to Venus and Mars, the manned mission potential of the solid core nuclear reactor had been exhausted. To extend manned flight over the rest of the solar system, more powerful propulsion systems were clearly needed. A large number of advanced nuclear propulsion systems had been studied theoretically and experimentally in the 60's and 70's, resulting in a shopping list for helionautical operations which appeared to be more extensive than it actually turned out to be (Table 1). The solid core reactor in its most advanced form (metal carbide, non-moderated) had been raised, by the end of the 70's, to a performance level of approaching 1000 sec specific impulse. In the fluidized-bed reactor, a concept originated by L. P. Hatch (1960-1), solid-fuel particles, suspended by viscous drag and their weight increased hundred to thousandfold by centrifuging, ride a high-velocity propellant gas stream, heating the gas as it passes. In the liquid core reactor, the temperature is increased to the point where the fuel has become molten in portions of the core and the gas is driven through it in bubbles. Both the fluidized bed and the liquid core reactor failed to offer a significant increase in specific impulse, in addition to a low thrust acceleration capability of the latter. A large variety of gaseous core reactor (GCR) systems were investigated, because the GCR offers a larger increase in specific impulse than the intermediate forms between the solid core and the gaseous core reactor. Research and technology work on some configurations was particularly successful, including the multi-cavity (multi-cell) reactor and the coaxial flow reactor. The GCR is the ultimate step in thermal propulsion, involving continuous operation and internal heating. The energy radiated into the surrounding solid walls and the reflector on the one hand, and the fundamental limitations in heat conduction and cooling on the other, provide the basic constraint for the specific impulse so attainable. While the upper theoretical limit lies in the region of 2500 to 3000 sec, the specific impulses achieved in practice lie in the 1800 to 2200 sec range. For missions in the 20,000 to 30,000 ft/sec velocity class, such as cislunar traffic, this provides a very economic gross payload fraction (60 to 70 percent). Even for Venus and Mars missions in the 40,000 to 60,000 ft/sec class, this still yields a gross payload fraction of 30 to 40 percent. However, for extended solar system missions, this specific impulse regime simply is not yet adequate.

Most of the major problems associated with the advanced solid core reactor, the fluidized-bed reactor, the liquid and the gaseous core reactors were recognized already during the early phases of investigation (cf. references 1960, 1961, 1962, 1963-5 through -13, 1964-5 through -9, 1965-4 and -5, 1966-2 and later years of the 60's). A survey of the expectations and problems recognized in those days is provided in the excellent reviews of references 1963-9 and 1966-2.

Table 1. Propulsion System Shopping List for Advanced Helionautical Operations at and Beyond 1990

ENGINE	SPECIFIC IMPULSE (sec)	THRUST ACCELERATION (g)	PROPELLANT
SOLID CORE REACTOR (GRAPHITE OR WATER MODERATED)	750 to 850	$\gtrless 1$	HYDROGEN
SOLID CORE REACTOR (METAL OR METAL CARBIDE; NON-MODERATED)	850 to 1000	$\gtrless 1$	HYDROGEN
FLUIDIZED-BED REACTOR	1000 to 1100	0.1 to 1	HYDROGEN
LIQUID CORE REACTOR	1100 to 1200	10^{-4} to 10^{-3}	HYDROGEN
GASEOUS CORE REACTORS	1700 to 2500	$\gtrless 1$	HYDROGEN
NUCLEAR PULSE			
FISSION PULSE UNITS	2500 to >5000	$\gtrless 1$	METAL
FUSION PULSE UNITS	5000 to >10,000	$\gtrless 1$	METAL
CONTROLLED THERMONUCLEAR REACTOR	10,000 to >100,000	10^{-3} to 10^{-4}	DEUTERIUM HELIUM-3
NUCLEAR ELECTROSTATIC	5000 to >30,000	$\sim 10^{-4}$	CESIUM MERCURY

This left essentially three contenders for the helionautics sector of astronautics: The nuclear pulse (NP) drive, the nuclear electric (NE) drive and the controlled thermonuclear reactor (CTR) drive, all of which attain specific impulses well in excess of 2500 sec. None of these concepts were new. They had been under study and experimentation for at least a quarter of a century, the concepts being even older.

Before discussing these concepts, it should be pointed out that the objectives of helionautical operations in the late 80's and the 90's were to reach the planets Jupiter and Saturn and their moons; and to establish a shuttle service capability between Earth and the planets of the inner solar system. Figure 22 shows the mission velocity-time correlation for increasingly fast round-trip missions to Venus and Mars. Table 2 shows representative round-trip mission velocities to Mercury and Jupiter. The influence of capture orbit ellipticity in the very strong gravitational field of Jupiter is apparent. However, if one of Jupiter's large (Galilean) moons is to be visited, the mission velocity, with benefit of capture maneuver in the gravitational field of one of these sizeable moons, is at least in the range of 110,000 to 125,000 ft/sec. While a certain relief (about 20,000 ft/sec) can be obtained in missions to Mercury by utilizing Venus fly-by opportunities, these are not always available. For Jupiter, a combination of Venus fly-by with a powered maneuver in the Venus gravitational field is helpful (reduction in mission velocity 20,000 to 40,000 ft/sec), but only for missions which are inherently more expensive than those considered above in which the returning vehicle enters the inside of the Earth's orbit only slightly or not at all. Table 3 shows the alternative mission profile options for round-trip missions to all planets; but each of these fly-by options is available only temporarily at certain time intervals. Consequently, the problem in 1980 was to decide how, in the late 80's and in the 90's, a helionautical mission capability of 80,000 to 120,000 ft/sec (instead of 40,000 to 60,000 ft/sec as in the second round) could be obtained.

In order to illustrate the importance of very high specific impulse for these helionautical missions, Figure 23 shows the variation of the required specific impulse versus overall mission velocity for given values of the vehicle's gross payload fraction λ. The range of gross payload fractions given on each line is determined by the ratio of hardware to propellant weight of the propulsion module. The gross payload fractions shown are based on a one-stage vehicle. For a two-stage vehicle[3)] the overall mission velocity for each line would double, but the gross payload fraction would be equal to the square of the values shown. The gross payload fraction is the ratio of overall payload mass carried during the mission to vehicle gross mass at the beginning of the mission. It is seen that for a gross payload fraction of 0.5 to 0.6 for missions to Venus and Mars and for missions to Jupiter and Mercury, the specific impulse should be in the

3) of equal specific impulse for both stages and comparable mass fraction

Table 2. Representative Round-Trip Mission Velocities to Mercury and Saturn

TARGET PLANET	MISSION MODE	ROUND-TRIP CAPTURE MISSION VELOCITY (10^3 ft/sec)
MERCURY	CIRCULAR ORBIT CAPTURE; $r^* = 1.1$ $T_1 = 80 - 110$ d $T = 330 - 350$ d	90 - 105 (UNRETARDED HYPERBOLIC ENTRY)[1] 100 - 110 (EARTH ENTRY VELOCITY 50,000 fps) 127 - 137 (CIRCULAR ORBIT CAPTURE, $r^* = 1.06$)[2]
JUPITER	ELLIPTIC ORBIT CAPTURE; $n = 3$; r_p^* $T_1 = 460$ d $T = 1000 - 1050$ d	130 - 135 (UNRETARDED HYPERBOLIC ENTRY)[1] 135 - 140 (EARTH ENTRY VELOCITY 50,000 fps) 162 - 167 (CIRCULAR ORBIT CAPTURE; $r^* = 1.06$)[2]
	ELLIPTIC ORBIT CAPTURE; $n = 30$; $r_p^* = 1.1$ $T_1 = 460$ d $T = 1000 - 1050$ d	75 - 80 (UNRETARDED HYPERBOLIC ENTRY)[1] 80 - 85 (EARTH ENTRY VELOCITY 50,000 fps) 107 - 112 (CIRCULAR ORBIT CAPTURE; $r^* = 1.06$)[2]

n = RATIO OF APOAPSIS TO PERIAPSIS

r_p^* = PERIAPSIS DISTANCE IN PLANET RADII

r^* = CIRCULAR ORBIT DISTANCE IN PLANET RADII

T_1 = EARTH-TARGET-PLANET TRANSFER PERIOD (d)

T = OVERALL MISSION PERIOD (d)

1) INTO EARTH'S ATMOSPHERE AT MISSION TERMINATION

2) ABOUT EARTH AT MISSION TERMINATION

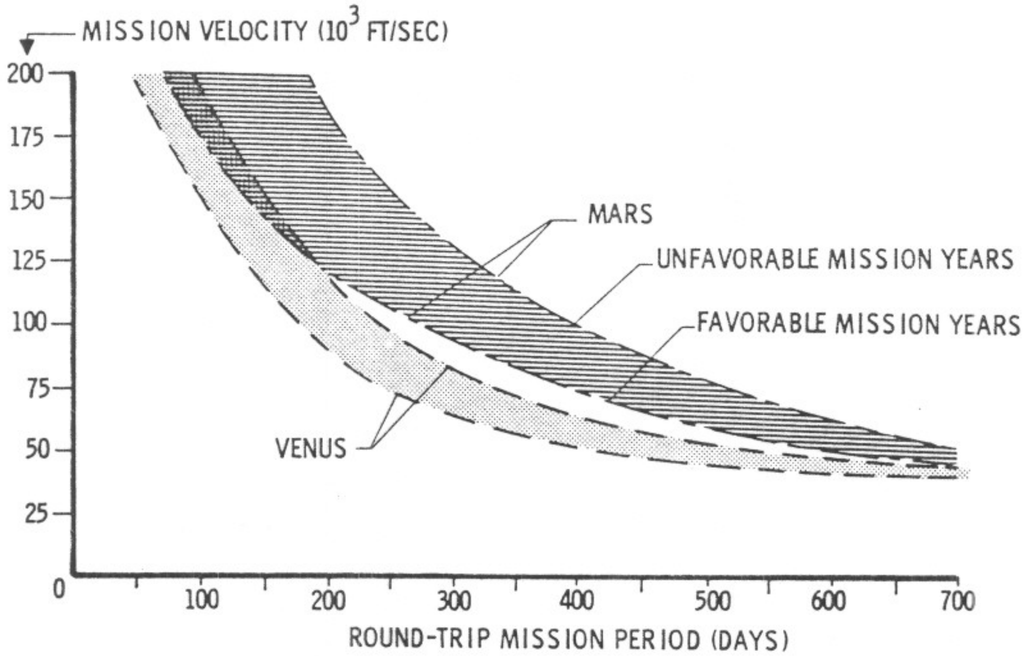


Figure 22. Velocity-Time Correlation for Round-Trip Missions to Venus and Mars Involving Capture Periods of 5 to 30 Days and Near-Earth Circular Capture at Mission Termination.

Table 3. Survey of Interplanetary Mission Profile Options for Flights to Planets of the Solar System

INTERMEDIATE MANEUVER

Table 3. Survey of Interplanetary Mission Profile Options for Flights to Planets of the Solar System

TARGET PLANET	INTERMEDIATE MANEUVER												
	OUTBOUND						EARTH-BOUND						
	NONE	FLY-BY OR SWING-BY					NONE	PERIHELION BRAKE	FLY-BY OR SWING-BY				
		Ve	Ju	Sa	Ur	Ne			Ve	Ju	Sa	Ur	Ne
MERCURY	*	*					*		*				
VENUS	*						*						
MARS	*	*					*	*	*				
JUPITER	*						*	* (*)	(*)				
SATURN	*		*				*	*		*			
URANUS	*		*	*			*	*		*	*		
NEPTUNE	*		*	*	*		*	*		*	*	*	
PLUTO	*		*	*	*	*	*	*		*	*	*	*

(*) IN THIS CASE THE PERIHELION OF THE RETURN ORBIT FROM JUPITER LIES ABOUT AT THE DISTANCE OF VENUS SO THAT THE PERIHELION BRAKE MANEUVER CAN BE CARRIED OUT SIMULTANEOUSLY WITH VENUS FLY-BY

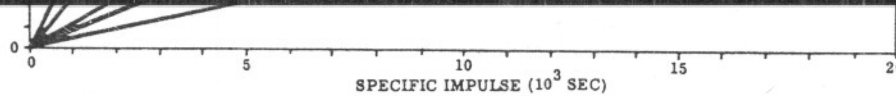


Figure 23. Variation of Required Specific Impulse Versus Overall Mission Velocity for Given Gross Payload Fractions.

range of 4000 to 9000 sec. Higher specific impulses are seen to pay off to a lesser degree. Lower values lead to a rapid decline in payload fraction; i.e., increasing transportation cost.

The nuclear pulse (NP) drive is based on the concept, originally advanced in 1945 by Dr. S. Ulam, Los Alamos Scientific Laboratory, of using nuclear explosives to provide the energy source (fuel) for high-performance space vehicle propulsion. In those days, and for many years thereafter, the nuclear fission or fusion explosive constituted the most potent, compact energy source known. As you know, today we are on the verge of controlling a still more concentrated energy source, since we are well along in the development of the first anti-matter aggregate. On the basis of energy alone, a fission explosive provides a specific impulse of at least 200,000 sec, a fusion explosive could deliver upwards of 400,000 sec. These specific impulses are excessively high for most missions, even most helionautical missions; moreover, use of the nuclear fuel in "pure" form causes many problems which even today are not solved in their entirety. The energy released by the explosion must, therefore, be transmitted to a propellant which is heated and which subsequently provides the thrust. The conventional method of doing this is to set off the nuclear explosive in a combustion chamber filled with propellant (e.g. water or other liquid material) which is being evaporated and subsequently ejected more or less diabatically through an expansion nozzle. This system, which we call internal nuclear pulse (INP), is very efficient inasmuch as all explosion products and the entire propellant mass are utilized for propulsion. The non-steady operation makes it possible for the system to reach higher peak pressures and temperatures in the combustion chamber, and therefore, a much higher specific impulse, than could be obtained in a steady-state operation. But the very fact that the explosion is internal imposes basic limitations on chamber pressures and temperature, hence on the specific impulse attainable. Therefore, while the INP had much to recommend it, the basically greater simplicity and superior growth potential of the external nuclear pulse (ENP) system won out over the superior efficiency of the INP, because the former advantages were more important in view of the enormous amount of energy available in nuclear explosives. In the ENP, not all the fuel and the propellant are utilized for propulsion; but the percentage which is utilized operates at a higher energy level and therefore can result in a superior performance. The ENP, or simply NP, operates on the basis of a large number of small nuclear explosive systems, called pulse units, which are stored in the vehicle. These pulse units are sequentially ejected and detonated external to and some distance behind the vehicle (Figure 24). A certain percentage of the expanding gaseous debris of each detonation, in the form of a high-density high-velocity plasma, is intercepted by the base of the vehicle, called a pusher plate. The momentum of the intercepted plasma is transferred rapidly to the pusher plate, moving the pusher forward at high acceleration. This acceleration is moderated by shock absorbing devices to levels of a few g's in the forward end of the vehicle, well within human tolerance. After compressing the shock absorbers the pusher returns to its natural position prior to accepting the subsequent pulse. The overall velocity increment received by the vehicle (Figure 25, 1964-10) depends upon the number of pulses and is,

Figure 24. External Nuclear Pulse Engine Concept

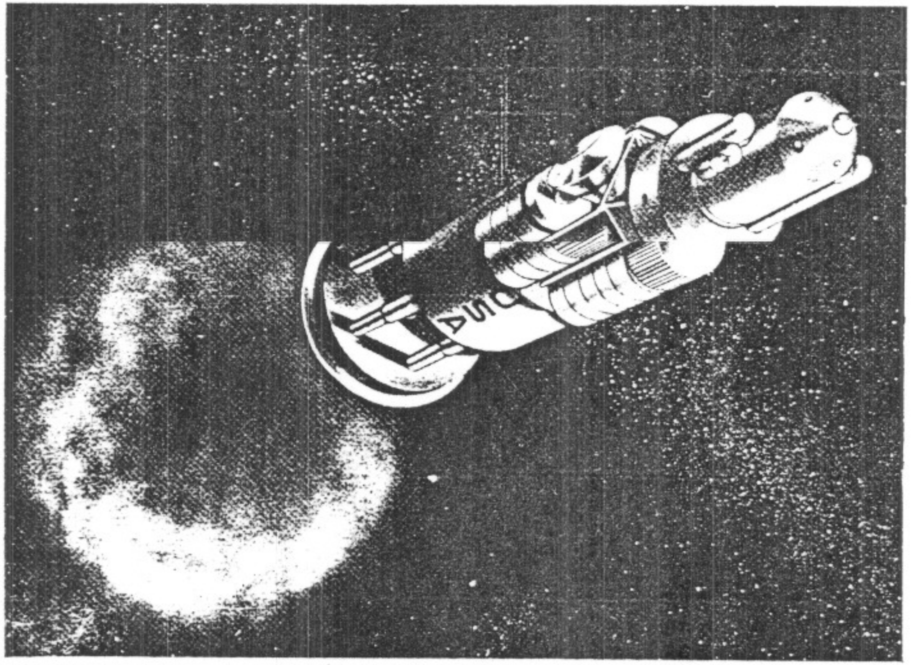


Figure 25. Nuclear Pulse Vehicle During Powered Maneuver

therefore, controlled by the number of pulse units expended in a given maneuver. Systematic research on this concept was initiated back in 1956 by Dr. T. B. Taylor of General Atomic Division of General Dynamics Corporation, and has been continued under the sponsorship of ARPA, USAF, NASA and General Dynamics for about nine years subsequently (1964-10; 1965-6), when it was brought to a temporary halt, to be resumed vigorously later when the usefulness of this propulsion system for missions (1965-7) caused it to be selected for ultimate development as one of the two principal helionautical drives. An early design carried out for NASA in 1964/65 envisioned a Saturn V compatible (by diameter) version with a specific impulse of 2500 sec, dry weight of the basic (non-fueled) propulsion module of 200,000 lb and with an effective thrust of 780,000 lb (1965-1). The nuclear pulse system in the original form used nuclear fission units which presented, for use in low earth orbit and in the radiation belt, a problem because of the various radioactive materials released with each explosion. The specific impulse of the NP system has meanwhile been increased to 10,000 sec, and in the future this figure will no doubt be doubled. The NP system could have been developed already in the 70's to be available for the first manned planetary mission in the early 80's. However, for that mission, a real development need could not be established in view of the commitment in the solid core nuclear engine development. Moreover, a political problem was associated with the drive, stemming from the fact that NP propulsion applies nuclear charges. The nuclear weapons test ban treaty then in force, seeking ultimately "the discontinuance of all test explosions of nuclear weapons for all times" excluded all nuclear explosions, except for underground tests. In that form, the treaty in effect prohibited the development, testing and operation of nuclear pulse vehicles. The treaty, however, provided procedures for its own amendment and its spirit was, of course, not to prohibit the development of advanced space propulsion and the exploration of space. These provisions eventually made it possible for the interested parties to obtain amendments which allowed the development of the NP system.

The CTR is a fusion propulsion system, based on the conversion of hydrogen to helium. In the 60's, Aerojet-General Nucleonics began to conduct for the Air Force controlled fusion research, under the direction of

- A proper magnetic geometry must be developed which permits a stable steady-state reaction condition.
- A lightweight method to generate and maintain the strong magnetic field necessary to contain the thermonuclear plasma must be devised; and "lightweight" implies in this case also low power requirement.

In the early 70's the practical feasibility of several self-sustained thermonuclear plasmas was demonstrated. These were essentially of three types: Deuterium-Deuterium (D-D), Deuterium-Tritium (D-T) and (Helium-3)-Deuterium (^{3}He-D) reactions. The D-D and the D-T reactions release large amounts of energy in the forms of neutrons (approximately 50 and 75 percent, respectively). For this reason, the CTR system suggested by Dr. John Luce, is based on the thermonuclear reaction of deuterium with a Helium-3 isotope to produce Helium-4 and protons, according to the equation:

$$D + {}^3He \longrightarrow {}^4He(3.6\ Mev) + H(14.7\ Mev).$$

Both reaction products are charged particles and can, therefore, be confined by magnetic fields, thereby isolating the plasma from the wall. A small amount of reactions of the type:

$$D + T \rightarrow {}^{4}He(3.5\ Mev) + n(14.1\ Mev)$$

and

$$D + D \begin{cases} \nearrow {}^{3}He(0.82\ Mev) + n(2.45\ Mev) \\ \searrow T(1.01\ Mev) + H(3.02\ Mev) \end{cases}$$

will occur also as the result of intermediate formations.

These secondary reactions, however, can be minimized so as to produce only a small fraction (1-2 percent) of the total system power in neutrons. The plasma is confined in a magnetic geometry which combines the cusp and the regular mirror geometry and which has both the needed resistance to instabilities and a long adiabatic confinement time (1961-4). The magnetic field is maintained by superconducting coils made of the intermetallic compound of niobium and tin. The superconductive characteristics of certain intermetallic compounds, originally discovered by J. E. Kunzler et al (1961-3), provided the breakthrough needed to develop a comparable lightweight, low-power system for generating and maintaining very strong magnetic fields equal to, and in excess of, 200,000 gauss without the heavy coils used earlier, the extensive power conversion system, large radiators, structural parts and the large electric power requirements. Among several alternatives, niobium-zirconium (Nb-Zr), niobium-tin alloy (Nb_3Sn) and vanadium-gallium (V-Ga), the last named was found to be most attractive in terms of critical magnetic fields at slightly higher cryogenic temperatures, in smallness of neutron cross-sections and in weight per unit of current carried. The principal "building blocks" of the CTR engine are (1963-16)

- Bremsstrahling Heat Shield
- Superconducting Coils
- Coil Support Structure
- Thermal Protection Sub-system
- Coolant Circulation Sub-system
- Radiation Shielding of Crew
- Cryogenic Subsystem

These building blocks are described briefly in Table 4. The fuel consists of ${}^{3}He$(60%) and D(40%). The reaction products are ${}^{4}He$ and H. The

Table 4. Survey of CTR Engine Sub-Systems

SUBSYSTEM	DESCRIPTION AND TYPICAL VALUES
o BREMSSTRAHLUNG HE:IIELD	STRUCTURE WHICH SURROUNDS THE PLASMA, IN GYROMAGNETIC RADIATIONS. THE SHIELD REF THESE RADIATIONS (WHICH CONSTITUTE A SIG GENERATED) BACK INTO THE PLASMA AND ABSO THE AMOUNT OF RADIATION ABSORBED, THE SH ACTIVELY COOLED WITH A LIQUID METAL COOL THE ABSORBED ENERGY IS USED FOR ELECTRIC OR ROTATING EQUIPMENT).
o SUPERCONDUCTING C	FABRICATION, SUPPORT AND COOLING OF SUPE TICAL IMPORTANCE IN CTR ENGINE DESIGN. POUND IS VANADIMN-GALLIUM ($Va_{2.95}$ Ga) (STRONG MAGNETIC FIELDS UP TO SEVERAL HUN ADEQUATE FOR CONTAINING THE FUSIONING PI THE COILS ARE SUBJECTED TO EXTREME COMPR MAGNETIC FIELDS. A SPECIAL STRUCTURE SU CRYOGENIC TEMPERATURES. THE SUPPORT STR INSULATED FROM THE COILS. BECAUSE OF CO INSULATION, THE COILS ARE UNDER CONTINUO TIONS. TYPICAL VALUES FOR THE THERMAL F THROUGH THE INSULATION ARE 15-17 KW FROM FROM CONDUCTION OR RADIATION TRANSFER. THE COILS AT TEMPERATURES BETWEEN 10 AND
o THERMAL PROTECTIO SUB-SYSTEM	THROUGHOUT THE SYSTEM, SUPERINSULATION LONG MEAN FREE PATH OF THE HIGH ENERGY N MAKES IT UNFEASIBLE TO TRY TO REDUCE THE POSITION OF SHIELDING MATERIALS BETWEEN CONDUCTING COILS. FOR A FIXED NEUTRON ABSORBED CAN BE REDUCED MOST EFFECTIVELY TRANSPARENCY", i.e. BY MINIMIZING THE MA GENERALLY USING MATERIAL WITH SMALLEST CRYOGENIC COOLANT IS GASEOUS HELIUM, BEC SECTION OF LIQUID He WOULD INCREASE THE

Table 4. (Cont)

Table 4. (Cont)

SUBSYSTEM	DESCRIPTION AND TYPICAL VALUES
o COOLANT CIRCULATION SUB-SYSTEM FOR COOLING AGAINST NEUTRON HEATING	THROUGHOUT THE ENGINE, A LARGE NUMBER OF MATERIALS ARE EXPOSED TO NEUTRON RADIATION AND HEATING ASSOCIATED WITH VARIOUS MECHANISMS OF NEUTRON-NUCLEI INTERACTIONS. A CTR ENGINE WITH A TYPICAL WEIGHT OF ABOUT 33,000 LB, CONTAINING TITANIUM (55%), ALUMINUM (40%), STEEL, VANADIUM, CHROMIUM AND GALLIUM, ABSORBS ABOUT 17 KW AT A FLUX OF ABOUT $3.6 \cdot 10^{11}$ OF 2.45 MEV-NEUTRONS PER CM^2 AND SECOND, AND ABOUT $1.09 \cdot 10^{11}$ OF 14.1 MEV-NEUTRONS PER CM^2 AND SECOND. COOLANT IS GASEOUS HELIUM.
o RADIATION SHIELDING	THE ONLY SECTION OF THE CTR ENGINE FOR WHICH NEUTRON TRANSPARENCY IS NOT ACCEPTABLE LIES IN THE DIRECTION TOWARD THE CREW SECTION, I.E. IN FORWARD DIRECTION. AN APPROPRIATELY THICK NEUTRON SHIELD MADE OF LITHIUM HYDRIDE AND PELYETHYLENE AT ITS FORWARD (COOLER) PORTIONS SEPARATES THE PLASMA FROM THE PAYLOAD SECTION AND PROVIDES APPROPRIATE SHADOW SHIELDING.
o CRYOGENIC SUB-SYSTEM	THE MASS OF THE CRYOGENIC SYSTEM CONSISTS PRIMARILY OF (A) CRYOGENIC SPACE RADIATORS, (B) THE POWER PLANT WHICH DRIVES THE CRYOGENIC HELIUM COMPRESSORS, (C) THE CRYOGENIC PLANT (COMPRESSORS, TURBINES AND RECUPERATIVE HEAT EXCHANGERS). THE GREATEST AMOUNT OF TECHNOLOGICAL ADVANCEMENT OVER THE 1970 STATE-OF-THE-ART WAS REQUIRED IN REGARD TO (C), PARTICULARLY WITH REGARD TO THE DEVELOPMENT OF HELIUM COMPRESSORS FOR LARGE COMPRESSION RATIOS, AND WITH REGARD TO THE MANUFACTURING OF LONG REGENERATIVE HEAT EXCHANGERS WITH MANY THIN TUBES OF SMALL DIAMETER. THE CRYOGENIC SUB-SYSTEM CONTINUOUSLY REMOVES THERMAL ENERGY FROM THE SUPERCONDUCTING COILS MAINTAINED AT A TEMPERATURE OF 10-20 DEG K.

propellant for thrust augmentation consists of deuterium (D_2) and hydrogen (H_2). A critical parameter in fusion plasma physics is the ratio β of expanding plasma pressure to the confining magnetic pressure. Depending upon plasma stability and the magnetic geometry, β can range between 0.1 (magnetic field pressure required in 10 times the plasma pressure) and 1.0 (both pressures equal). It is obvious that $\beta = 1.0$ requires a highly stable plasma and a suitable magnetic geometry. Some of the early plasma machines (stellarator) in the 50's achieved β-values which were limited to even less than 0.1. However, some cusp machines achieved β-values of 0.8. In the early 70's, β-values of about 0.2 were attained in a predictable and reliable way. Later β-values climbed to 0.4, 0.6 and 0.8. The correlation between β and the optimized motor weight, in terms of lb per kw power in the thrust-producing exhaust jet, is shown in Figure 26 (1963-16; 1966-3). The values are shown for a range of kinetic energies of the electrons in the plasma corresponding to an upper and lower limit of the electron temperature T_e; and for two thrust values, namely, 28 and 100 megawatt. It is seen that most of the gains in power-specific weight are attained when β has reached a value of 0.4 to 0.5.

To achieve ignition, high-energy neutral particles are injected into the center of the magnetic geometry. With intense magnetic fields, the neutral particles injected are ionized and trapped. By injecting and trapping a high-energy (^{3}He-D) plasma into the magnetic field, the needed particle density can be quickly built up to the point where ignition takes place. Once ignition of a self-sustained fusion reaction has been achieved, the needed fuel is fed into the reacting plasma region at low energy, where it is ionized and heated to the required temperature. With continuing reaction, the ions in the plasma are finally scattered into the escape cone of the confining mirror-type field. By adjusting the magnetic strength of the rearward mirror coil to be slightly weaker than the forward mirror, the plasma is expelled in the aft direction. The system can be operated uninterruptedly for several years.

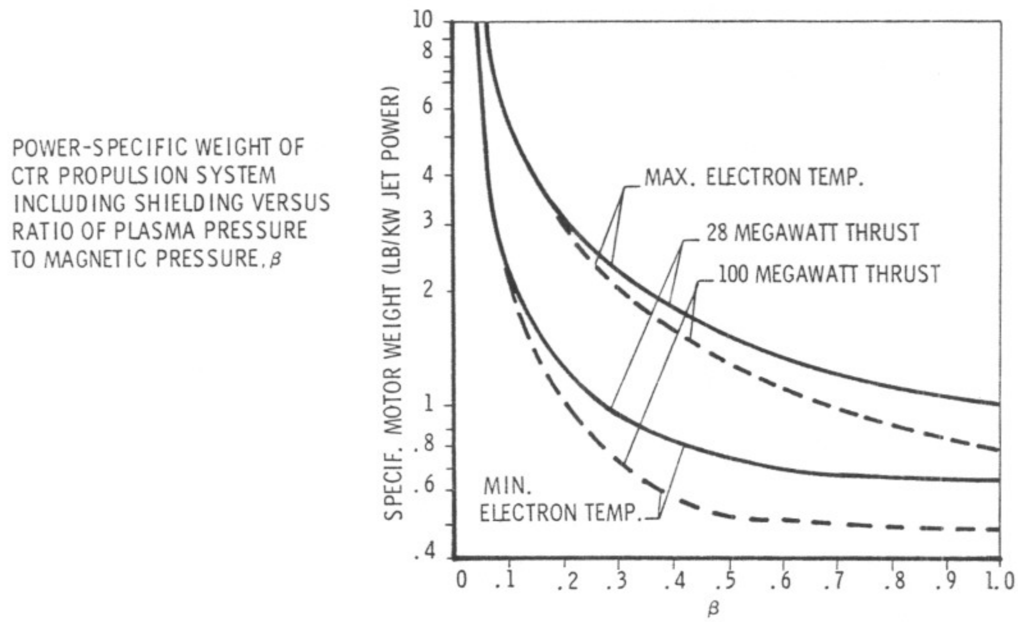


Figure 26. Power-Specific Weight of CTR Propulsion System Including Shielding Versus Ratio of Plasma Pressure to Magnetic Pressure

The attainable specific impulse is given by the simple relation

$$I_{sp}(\text{sec}) = \frac{45,910}{F(\text{lb})}\ P_j(\text{megawatt})$$

$$= \left[\frac{45,910}{\dot{w}(\text{lb/sec})}\ P_j(\text{mw})\right]^{1/2}$$

where P_j is the power contained in the exhaust jet, F the thrust and $\dot{w}$ the quantity expelled in the exhaust jet. The equation shows that, for a jet power P_j = 100 mw and a thrust of 100 lb, the specific impulse is 45,910 sec. Even at 1000 lb thrust, the specific impulse is still almost 4600 sec. For a jet power of 100 mw, the fuel (D + ^{3}He) consumption is about 1.14 lb/day or $1.3 \cdot 10^{-5}$ lb/sec at a specific impulse of about 600,000 sec and a thrust of approximately 7.7 lb. Thrust and specific impulse can be varied by mixing the exhaust gas with additional cold propellant (D_2, H_2)

in a thrust augmenter. The above numbers give an indication of the wide range of thrust and specific impulse which is comparatively readily available to the pilot of the CTR. Suppose, a CTR powered interplanetary vehicle is ready for departure in a near-Earth orbit (1.5 hrs period of revolution) at an orbital departure weight of 10^6 lb. At a tangentially applied constant thrust of 1000 lb, and a specific impulse of nearly 4600 sec, it takes approximately 106 times the original period of revolution or 26.5 hours (at 790 lb/hr propellant consumption) to reach local parabolic velocity at about 28 Earth radii distance. At this point, energetically speaking, the space vehicle has escaped the Earth's gravity field and henceforth can operate within the framework of the heliocentric gravitational field. At Earth distance, the Sun's field has a strength of $6 \cdot 10^{-4} g_{Ea}$ (g_{Ea} = gravitational acceleration at Earth's surface); this means at about 28 Earth radii, the thrust can be throttled to essentially heliocentric conditions. The vehicle has consumed roughly 210,000 lb propellant. It's weight is, therefore, 790,000 lb and its instantaneous acceleration approximately $1.27 \cdot 10^{-3}$g. Cutting the thrust to one-fifth of its original value, reduces the instantaneous acceleration to $2.5 \cdot 10^{-4} g_{Ea}$ (resulting in the fairly high heliocentric acceleration of about $0.4 g_{\odot}$, hence, a rather fast heliocentric powered flight path), reduces the propellant consumption to about 12.4 lb/hr (or about 300 lb/day), and raises the specific impulse to some 24,000 sec. Continuing at this thrust, accelerating tangentially along a flight path which leads away from the Sun, the space vehicle reaches parabolic velocity with respect to the Sun at a distance of slightly less than 1.1 AU, in less than 60 days and at a propellant consumption of 18,000 lb. The pilot could now throttle the motor back to "idling", i.e. no thrust augmentation, hence, no significant further acceleration, no propellant consumption and negligible fuel expenditure. At heliocentric parabolic speed, the space vehicle crosses the Mars orbit some 60 days later and the Jupiter orbit some 300 days later. For capture at any of these planets, of course, a heliocentric braking maneuver is required such that the vehicle approaches the planet at near-parabolic velocity which subsequently is reduced within the planet's g-field, at increased thrust, to the value for the desired capture orbit. The above data do not describe an optimized flight profile but serve to illustrate the fact that the CTR has a flexibility and power for helionautical flights which rivals those of a supersonic turboramjet aircraft for terrestrial flights. Figure 27 shows a CTR vehicle concept (propulsion section, with minor modifications of the cryogenic radiator configuration, based on reference 1963-16) for P_j = 100 mw thrust power.

The concept of space vehicle propulsion by means of electrically expelled charged particles seems to have been suggested first by R. H. Goddard in 1906. The concept received the first extensive treatment by H. Oberth in 1929. Various other preliminary evaluations and discussions followed. In 1954/55, the first comprehensive study of an engineering concept of an electrostatic (ion) engine and its application to a manned Mars mission was presented by E. Stuhlinger (1955-1) which stimulated many of the subsequent engineering studies of this system. His concept was based on the ionization of alkali atoms at hot surfaces of platinum or tungsten, due to the fact that the work function of these metals is greater than the ionization energy of the alkali atoms. Stuhlinger found that a combination of cesium and tungsten appeared to be most attractive. Two other methods which have been developed successfully are the plasma ion source and the bombardment ion source. In the first method, a plasma is generated by means of electric arcs. The ions are extracted by means of an electric field and concentrated into a more or less dense beam by a magnetic field (1957-3). In the bombardment method, ions are produced by the collision of high speed electrons with atoms.

In all cases, the electrostatic drive needs an original power source to provide the electric field in which the ions are accelerated to high exhaust velocity and which provides the energy for heating the tungsten surface (for re-evaporation of the cesium ion), for energizing the electric arc or for generating the electron source. The above equation for the attainable specific impulse is valid also for this drive. Writing it in the form of original power required for generating 1 lb of thrust,

$$P_o(\text{mw/lb F}) = \frac{I_{sp}}{45{,}910\ \varepsilon_c\ \varepsilon_t}$$

where P_o is the power at the source (e.g. the nuclear reactor), ε_c is the conversion efficiency to electric power P_e(emw) and ε_t is the conversion

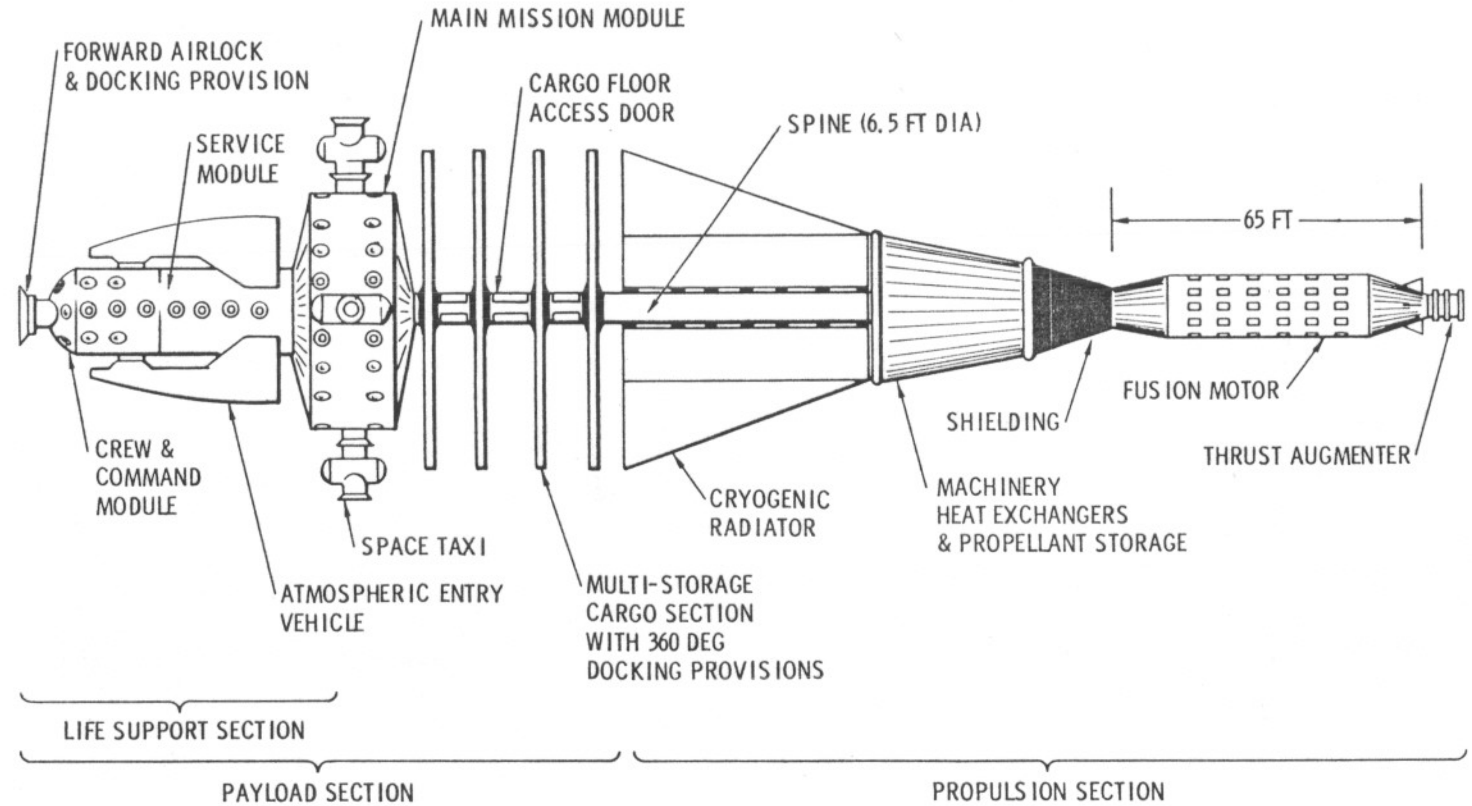


Figure 27. Controlled Thermonuclear Reactor Vehicle Concept

of the electric power to jet power P_j (mw), that is, the kinetic energy of the ion jet leaving the engine. The efficiency of turbo-electric and thermionic conversion systems grew from 20 to 40 percent in the last 30 years (i.e. $\varepsilon_c = 0.2$ to 0.4). The thrust conversion efficiency ε_t grew from 0.65 to 0.95, raising the product $\varepsilon_c\ \varepsilon_t$ from 0.13 to 0.38, or the term in the denominator of the above equation from approximately 6000 to 17,400. It means that the generation of 1 lb of thrust with $P_o = 1$ mw yielded a specific impulse of 6000 sec, and in the course of time, was raised to 17,400 sec; or, conversely, the propellant consumption was reduced from 0.6 lb/hr to 0.207 lb/hr for 1 lb thrust and 1 mw original power. At a conversion efficiency of 0.4, it follows that 600 kw of thermal energy must be removed by means of space radiators for every megawatt original power. Because of the mass of the conversion system and of the radiator, the power-specific weight of this power generation system could only with considerable effort and for very large systems be reduced to 5 lb/ekw. Weights of this order can be reduced significantly only with magnetohydrodynamic (MHD) generators (1958-4, 1959-9,-10, 1961-7); but this leads to another type of electrical propulsion which culminates in the CTR drive.

It is important to note, however, that aside from the NP and the CTR, the nuclear-electrostatic (NE) drive is the only one offering specific impulses in the 5000 to 20,000 sec regime. It must also be remembered that in the 60's and part of the 70's the NP and the CTR had great difficulty in competing with the NE concept. The development of the NP was inhibited by the atomic test ban treaty. Even if it could have been developed, it would have had to be based on Earth orbital departure and its pulse units would have had to be detonated while passing through the radiation belt. While it was shown that the effect of contamination of the belt, especially by injecting masses of electrons, could have been minimized by launching from polar orbits, even the remaining unavoidable perturbation of the belt would have caused considerable opposition by scientists and concern among space engineers who were populating the belt regions with growing numbers of surveillance and other utilization satellites, on whose reliability an increasing portion of the world population began to depend for security, safety and service; and who were also injecting manned space systems into belt regions. Today, we have inexpensive

Earth-to-orbit and cislunar transportation and we operate a Lunar launch facility. These factors, and the high-energy missions not considered in those days, make the use of the NP drive from and to the Moon both attractive and economical; a situation which is far different from the one 25 to 30 years ago. While the NP concept even in those days did not seem to warrant serious doubts in its practical engineering feasibility, it must be noted that this could not, to the same degree, be said of the CTR drive. It is true that by the middle 60's this drive had been made conceptually feasible by the development of magnetic trapping, new metallic superconductors and high-energy neutron injection. But the problems of plasma stability, of ignition and self-sustaining reaction of a fusion plasma and the engineering problems associated with the development and operation of the helium cryogenic system, posed questions which were considerably more severe than the problems recognized in conjunction with the development of the NE drive. On the basis of these facts, there existed, some 25 to 35 years ago, a considerable incentive to pursue the development of the NE drive as the comparatively most accessible road to attaining very high specific impulses.

By 1980, the situation had changed considerably. The political obstacles in the way of development of the NP were removed. The CTR had caught up with the technological head-start which the NP had possessed 10-15 years before; and it could be compared on the basis of equality, as to practical feasibility, with the NE drive which, as it became increasingly apparent, the CTR seemed to be able to outrun in terms of performance during the 80's, if given comparable support.

Comparing the CTR with the electrostatic NE system, it is found that the weight of the fuel and propellant needed for the CTR system is markedly lower, because, in contrast to the conventional power conversion systems (even thermionic generators), the lighter weight fusion system has a considerably higher optimum specific impulse. Moreover, the process of direct expulsion of exhaust from the thermonuclear reactor without an intermediate heat transfer system, as required in the NE drive, provides greater simplicity and higher reliability. A much smaller radiator surface is required, because fewer conversion processes are involved. This contributes to the relatively light weight of the CTR drive and reduces its vulnerability to meteorite damage, compared to the NE drive. Lower shield weight (fewer neutrons produced than in a uranium fission reactor of comparable size) and absence of poisonous radiation add to the low weight and operational advantages. Thrust and specific impulse can be varied in wider limits than with the NE drive. For these reasons, the choice between NE and CTR devices for interplanetary missions finally fell to the CTR.

A comparison of the NP and CTR shows that in terms of cost effectiveness (not initial cost, however) and specific impulse, the CTR is superior; but that in terms of ruggedness and thrust acceleration range, the NP is superior. Moreover, the NP not only can descend to, or ascent from any planet, so far as overcoming gravitational pull is concerned, but the NP is capable of operating in any atmosphere as well as in space. In fact, the NP is the only rocket engine which tends to work better in an atmosphere than in space, because it uses the atmospheric gases as propellant. The NP is therefore more suitable for "rough and dirty" missions where extremely hostile environmental conditions as well as the need for much thrust power are involved. Mission examples are:

- Descent to the surface of Venus
- Entry into or fly-through of comet heads
- Entry into the atmospheres of the Jovian planets
- Penetrating difficult environments, such as the asteroid belt and possibly "dirty and dusty" regions surrounding the major planets
- It should also be noted that by being able to enter the heads of comets, it appears to be possible to use the solid matter as a protective shield for trips very close to the Sun.

The CTR, being less rugged than the NP drive, is more suitable for missions into the inner solar system, that is into relatively cleaner regions of heliocentric space. It must stay in satellite orbits at the target planet or can attach itself to small moons, but it cannot descend. Flight in the inner solar system does not necessarily mean that the enormous specific impulse range of the CTR cannot be fully utilized. For example, being "restricted" to rendezvous with those asteroids which enter the inner solar system while passing through the perihelion requires a very large amount of velocity change (usually upwards of 100,000 ft/sec). The procurement cost per pound of propulsion hardware of the CTR is about 5 times higher than that of the NP. But the operational cost of the CTR is less expensive, since the system uses deuterium and helium instead of nuclear material and high density metallic propellant. With its operational cost effectiveness being superior and its specific impulse high, the CTR is particularly suitable as a heavy load carrier, such as is involved in exploiting metal ore or other resources on another celestial body. Typical mission examples are

- Mercury capture
- Venus capture
- Mars moons
- Asteroid hunting outside the asteroid belt
- Attachment to asteroids for protection can be imagined while traveling through "dirty" regions of space (Figure 28). In this fashion, the CTR can reach "cleaner" regions in the outer solar system beyond Jupiter and continue flight to the planets Saturn through Pluto and eventually into interstellar space.

For these reasons, the NP and the CTR emerged as the selected drives for solar transportation and by 1981 active development of both types was underway.

In the manned planetary sub-program, the national goals were based on the availability of both the NP and the CTR drive, by 1986-1988. They included:

(1) Establishment of a Solar Physics Station on Mercury
(2) Establishment of a Research and Supply Station on the Mars Moon Phobos for the purpose of supporting a Biological Research Station on the surface of Mars
(3) Manned Landing on Venus
(4) Comet Encke fly-through mission
(5) Establishment of a Research Station on one of Jupiter's Galilean Moons
(6) Establishment of a Research Station on the Jupiter Moon Titan.

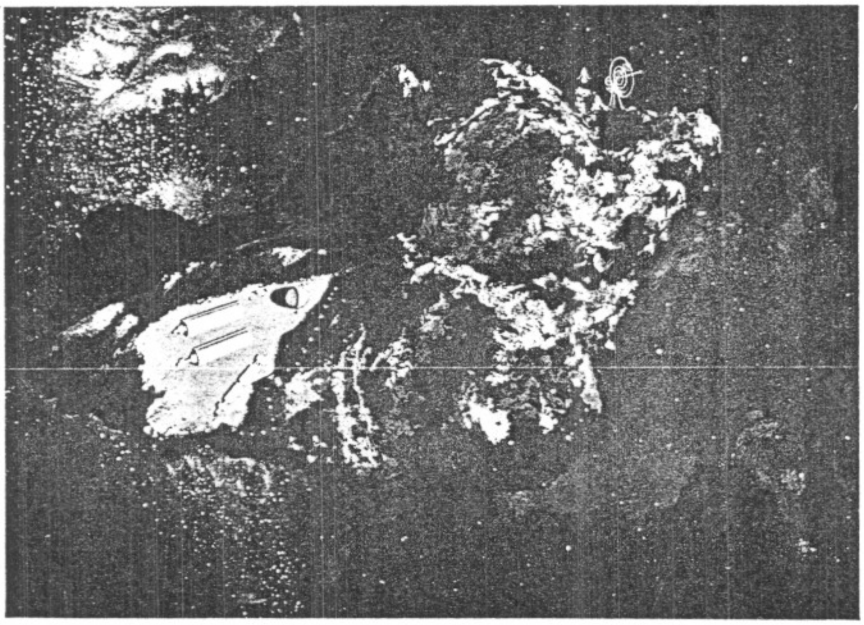


Figure 28. Interplanetary Vehicle Convoy Attaching Itself to an Asteroid for Better Protection During Passage Through a Portion of the Asteroid Belt. Suitable Section of Asteroid is Smoothed with Plastic Foam and Covered with a Surface Spray Which Hardens Rapidly.

For the unmanned planetary probe program the emphasis was placed upon two major objectives:

(1) Deployment of the Outer Solar System Voyager (OSSV) for the exploration of Jupiter and Saturn sufficiently in advance of the manned missions

(2) Development of a first-generation of trans-Pluto and interstellar probes.

The Evolution of Space Flight to FY 2001

Figure 29 presents an overview over the three principal manned space flight sub-programs and the manned deep space sub-program, each with a breakdown by its major project group.

Figure 30 shows the same evolution but in greater detail, by principal mission milestones. These correspond to the established national goals in the 1970-1985 and 1985-2001 time period.

The manned orbital, Lunar and planetary programs are based on the launch vehicle development shown in Figure 31. The upper branch refers to the family of personnel transports, the lower to the family of heavy payload launch vehicles. Following the development of Saturn V, three major development steps were taken in the family of heavy payload launch vehicles: uprating of Saturn V; a single-stage reuseable O_2/H_2 post-Saturn launch vehicle with attachable solid propellant rockets to provide variable payload capability; and, in the 90's the development of a magnetohydrodynamic propulsion system for a heavy payload vehicle whose delivery capability was extended beyond Earth's orbit, to reach Lunar satellite orbits in direct flight and return to the Earth's surface in regular shuttle operation. The uprating of Saturn V provided the necessary time to thoroughly plan and research experimentally, the difficult task of developing the post-Saturn vehicle. It finally was decided to provide a minimum payload of 700,000 pounds and a maximum payload of 1,400,000 pounds. This made it possible to cover with one to three launches the required orbital-departure mass requirements for planetary missions throughout the solar system, provided NP and CTR were the principal drives for the interplanetary vehicles. A smaller payload would have unduly increased the required number of launchings for the distant planetary missions and the associated orbital operations. A larger vehicle would have been launched so infrequently that the effort to make it reuseable would not have paid off for many years. The reuseable orbital transport began as a two-stage chemically proven system; thereafter, the first stage was replaced by an airbreathing hypersonic flight stage and finally the first stage was removed altogether, the second stage was converted to a single MHD driven system, powered by a solid core reactor, and chemical propulsion for lift-off and ascent to minimum permissible altitude for ignition of the nuclear engine.

In addition, and partly in support of the launch vehicle evolution, the orbital, Lunar and helionautical operations were supported by broad advanced research and technology programs which constituted the bedrock of the vast operational programs. In order to survey this program, the evolution of the space vehicle anatomy to FY 2001 is traced in Table 5 through the three time phases, comparing the space vehicle (generically speaking) to the human organism.

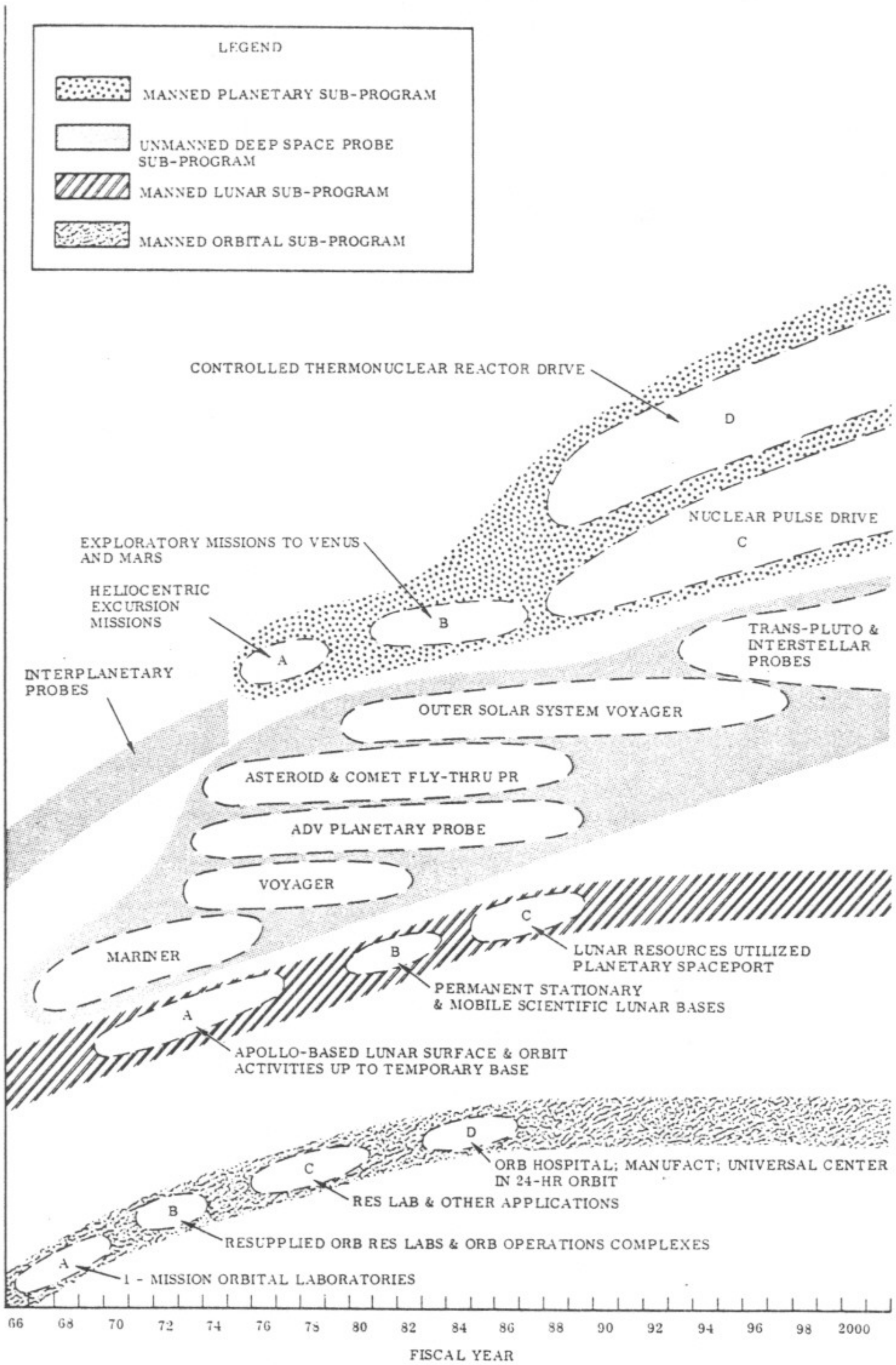


Figure 29. Evolution of Space Flight to F.Y. 2001 - Overview

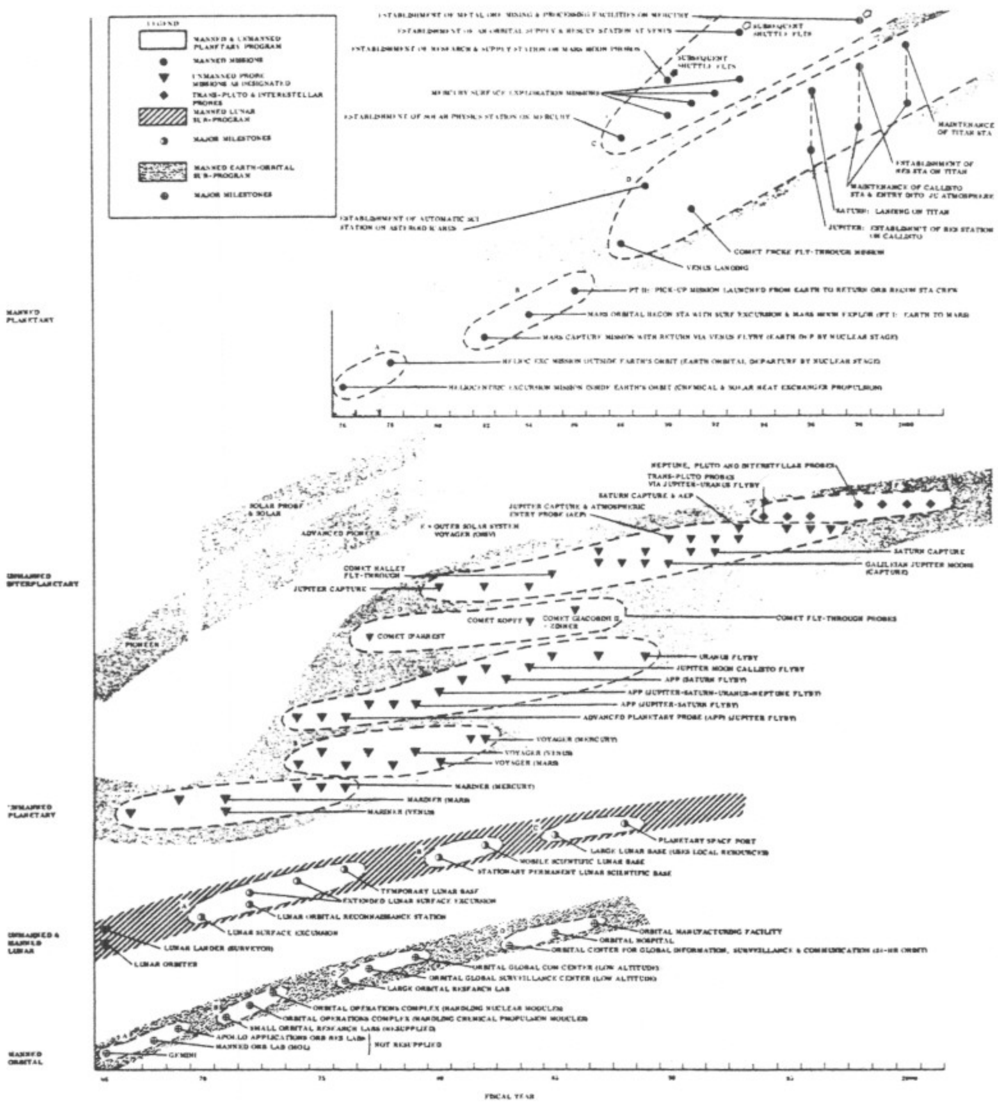


Figure 30. Evolution of Space Flight to F.Y. 2001 - Principal Mission Milestones

Table 5. Evolution of Space Vehicle Anatomy to Fiscal Year 2001 (Part 1 of 4)

TYPE	BIOLOGICAL SUBSYSTEM	TECHNOLOGICAL SUBSYSTEM	TIME PHASES :	1960-1970	1971-1985	1986-2000
(1) POWER & LOCOMOTION	MUSCLE	PROPULSION	CHEMICAL	$I_{sp} = 430$ sec	$I_{sp} \leq 450$ sec	$I_{sp} \leq 600$ sec
			SOLAR HEAT EXCHANGER		$I_{sp} \leq 700$; $F \sim 10$ lb	I_{sp} 900; $F \sim 50$ lb
			NUCLEAR SOLID CORE REACTOR		$I_{sp} \leq 800$; $F \sim 250k$ I_{sp} 900; $F \sim 50k$	$I_{sp} \leq 950$; $F \sim 250k$
			NUCLEAR ELECTRIC		$I_{sp} \leq 5000$ $F \sim 0.01$-1 per engine	$I_{sp} \leq 20,000$ $F \sim 5$ per engine
			NUCLEAR-POWERED MAGNETOHYDRODYNAMIC GENERATOR PROPULSION			$I_{sp} \leq 2000$ (SPACE) ≤ 5000 (AIR AUGMENTED) $F \sim 250k$ (SPACE) $\sim 700k$ (AIR AUGMENTED)
			NUCLEAR PULSE			$I_{sp} \leq 20,000$ $F \sim 10^6$
			CONTROLLED THERMONUCLEAR REACTOR DRIVE			$I_{sp} = 10^5$ to $5 \cdot 10^3$ $F = 50$ to 500
		ELECTRIC POWER GENERATION	PHOTOVOLTAIC	< 1 ekw[1]	1-5 ekw	>5 ekw
			FUEL CELLS	1-10 ekw	10-100 ekw	10-100 ekw
			THERMOELECTRIC	< 100 watt (RADIOISOTOPE (RTG))	0.1-1 ekw & SOLAR (STG)	>1 ekw RTG & STG
			NUCLEAR-DYNAMIC		≤ 60 ekw	≤ 60 ekw
			SOLAR-THERMIONIC		≤ 300 ekw	≤ 500 ekw
			NUCLEAR-THERMIONIC		≤ 1 emw	≤ 10 emw
			NUCLEAR-MAGNETOHYDRODYNAMIC		≤ 10 emw	≤ 10,000 emw
				[1] ekw = ELECTRICAL KILOWATT [2] emw = ELECTRICAL MEGAWATT		
(2) SUPPORT & STABILITY	SKELETON	STRUCTURES	MATERIALS	• TITANIUM ALLOYS	• BERYLLIUM-BORON COMPOSITES • SPACE RESISTANT PLASTICS • CRYOGENIC ELASTOMERS	ULTRA-PURE METALS

Table 5. (Part 2 of 4)

TYPE	BIOLOGICAL SUBSYSTEM	TECHNOLOGICAL SUBSYSTEM	TIME PHASES:	1960-1970	1971-1985	1986-2000
			DESIGNS	HONEYCOMB (GLASS REINFORCED RESINS ETC)	• FLEXIBLE RIGIDIZABLE STRUCTURES • WOVEN STRUCTURES • PRESSURE-STABILIZED STRUCTURES (RIGID OR FLEXIBLE)	SUPERCONDUCTING CRYOGENIC STRUCTURES WITH HIGH-INTENSITY MAGNETIC FIELD GENERATING CAPABILITY
(3) PROTECTION	• BONES • SKIN • HAIR • FAT	• HEAT PROTECTION ATMOSPHERIC ENTRY		PHENOLIC NYLON	• GRAPHITE COMPOSITES • REFRACTORY METALS • TRANSPIRATION COOLING	BLUNT BODIES AT PARABOLIC & HYPERBOLIC ENTRY VELOCITIES LIFTING BODY ENTRY AT CIRCULAR TO HYPERBOLIC VELOCITIES
		SOLAR HEATING		REFLECTIVE COATING	• STAND-OFF HEAT SHIELD SYSTEMS • ACTIVE REFRIGERATION SYSTEMS	ADAPTIVE COATING MATERIALS OF SELECTIVE SPECTRAL RESPONSE CAPABILITY
		WASTE POWER HEATING		RADIATORS	RADIATORS	RADIATORS
		• METEOROID PROTECTION		INTEGRATED METEOROID PROTECTION STRUCTURE	COMBINED SOLAR HEAT SHIELD AND METEOROID PROTECTION STAND-OFF SYSTEM	METEORITE HARDENED SURFACES AND RADIATORS
		• CORPUSCULAR RADIATION PROTECTION		MASS-SHIELDING (POLYETHYLENE, HYDROGEN-RICH FLUIDS (H_2O, CH_4, N_2H_4) AGAINST PROTONS; LEAD LINING AGAINST BREMSSTRAHLING FROM ELECTRONS)	MASS SHIELDING	FIELD-SHIELDING (DEFLECTION OF CHARGED PARTICLES IN PROTECTIVE MAGNETIC FIELD SURROUNDING THE SPACE VEHICLE)
		• SPACEWEAR FOR ASTRONAUTS		PRESSURE SUITS	ADVANCED PRESSURE SUITS	ASTROSKIN (SKIN ATTACHED LINER WITH MINUSCULE PRESSURE VESSELS FOR SUPPORT AND TEMPERATURE CONTROL; 360° VIEW HELMET MADE OF OPTICALLY ADAPTIVE MATERIAL; MICROMINIATURIZED O_2-REGENERATION & POWER PACKAGE FOR INDEFINITE OPERATING TIME)

Table 5. (Part 3 of 4)

Table 5. (Part 3 of 4)

TYPE	BIOLOGICAL SUBSYSTEM	TECHNOLOGICAL SUBSYSTEM	TIME PHASES:	1960-1970	1971-1985	1986-2000
(4) SENSORS & ORIENTATION	• EYES ($4.2 \cdot 10^6$ bits/sec)	• INFORMATION ACQUISITION (END-INSTRUMENTATION)				
	• EARS (8000 bits/sec)					
	• TOUCH SENSITIVITY	• FIELD SENSORS (MAGNETIC)	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
	• TEMPERATURE SENSITIVITY	• ELECTROMAGNETIC RADIATION SENSORS GAMMA RAYS X-RAYS ULTRAVIOLET VISIBLE INFRARED RADIO/RADAR	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
	• SMELL/TASTE					
		• CORPUSCULAR RADIATION SENSORS	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
		• DUST & MICROMETEOROID SENSORS	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
		• ACCELERATION/VIBRATION SENSORS	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
		• ACOUSTICAL SENSORS	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
		• TEMPERATURE SENSORS	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
		• PRESSURE SENSORS	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
		• FOREIGN MATTER ANALYZER (INORGANIC AND ORGANIC)	INCREASING SENSITIVITY & RANGE; DECREASING SIZE, WEIGHT & POWER REQUIREMENT			
		• GUIDANCE & NAVIGATION				
		• INERTIAL GUIDANCE		MTBF[1] $\sim 10^5$ hrs POWER $\sim$ 50 watt	MTBF $\sim 10^7$ hrs POWER $\sim$ 5 watt	MTBF: INDEFINITE POWER $<$ 5 watt
		• STELLAR GUIDANCE				
		• NAVIGATION METHOD		BY POSITION DETERMINATIONS	BY POSITION DETERMINATIONS	BY DIRECT MEASUREMENT OF VELOCITY COMPONENTS RELATIVE TO CENTER OF ATTRACTION
		• ARTIFICIAL-G PROVISIONS	NONE	• VEHICLE ROTATION • CENTRIFUGE		LIFE SUPPORT SYSTEM ROTATION

1) MTBF = MEAN TIME BEFORE FAILURE

Table 5. (Part 4 of 4)

TYPE	BIOLOGICAL SUBSYSTEM	TECHNOLOGICAL SUBSYSTEM	TIME PHASES:	1960-1970	1971-1985	1986-2000
(5) INTERNAL DISTRIBUTION, EXCHANGE & ELIMINATION	• NERVES	• CIRCUITRY		• INTEGRATED • THIN FILM	• METAL-OXIDE SEMI-CONDUCTORS	• CRYSTAL MATRIC CIRCUITRY • MOLECULAR CIRCUITRY
	• BLOOD CIRCULATION	• HEAT EXCHANGER SUBSYSTEMS				
	• HEART	• HYDRAULIC SUBSYSTEMS				
	• LUNGS	• PNEUMATIC SUBSYSTEMS				
	• STOMACH	• ECOLOGICAL SUBSYSTEMS		INORGANIC-OPEN	SEMI-CLOSED	INORGANIC-ORGANIC-CLOSED
	• KIDNEYS/BOWELS	• WASTE ELIMINATION SUBSYSTEM		• CENTRIFUGAL COLLECTORS • FILTERS	• CRYOGENIC COLLECTORS • OXIDATION, FLUORIDATION	
(6) REGULATORY CONTROL, SAFETY DEVICES & PROCESSES	• HORMONES	• PRESSURE CONTROL (VENTING) • THERMAL CONTROL				
	• VITAMINS	• INTRA-VEHICULAR CHECKOUT & DIAGNOSTIC DEVICES				
	• TRANSPIRATION	• FAIL-SAFE PROVISIONS		REDUNDANCY	MODULARIZATION REPAIRABILITY	SELF-REPAIR CAPABILITY
	• VESTIBULARY APPARATUS (EQUILIBRIUM SENSOR)	• ATTITUDE CONTROL	POINTING ACCURACY	1 min (ACTIVE) 3 deg (PASSIVE)	0.01 arc sec (ACT.) 1 deg (PASSIVE)	0.001 arc sec (ACTIVE) 1 min (PASSIVE)
		• STERILIZATION			• STERILIZATION OF SPACECRAFT ON EARTH • STERILIZATION IN SPACE AGAINST ALIEN LIFE FORMS	• AUTOMATIC STERILIZATION OF SUBSYSTEMS CARRIED BY INTERPLANETARY SPACECRAFT
(7) INTELLIGENCE	• BRAIN	• INFORMATION HANDLING	OPERATIONS PER SEC	~75,000	~ 500,000	~3,000,000
			MEAN TIME BEFORE FAILURE (hrs)	~15,000	~100,000	INDEFINITE
			POWER REQUIREMENT (watts)	~100	~ 10	3-6
		• COMMUNICATIONS		RADAR	RADAR & LASER	MOSTLY LASER
			POWER (kw)			
			UNMANNED PLANETARY	0.1	2	5-10
			MANNED PLANETARY		2-5	10-100
			DATA STORAGE (Bits) UNMANNED PLANETARY	10^7 (MARS)	10^8-10^{14}	10^{15}-10^{20}
		• DEEP SPACE COMMUNICATION NETWORK		DSIF (NASA/OSSA) CLEARWATER (NASA/MSFC)	SAME; IMPROVED	LUNAR DEEP SPACE COMMUNICATION NETWORK
		• CREW AND MANNED VEHICLE CONTROL CENTER		DEVELOPMENT IN ORBIT	DEVELOPMENT IN ORBIT USE IN SPACE STATIONS, CISLUNAR & PLANETARY VEHICLES FOR UP TO 600 DAY MISSION PERIOD	USE OF CONTROL CENTERS IN VERY DEEP SPACE VEHICLES, SHUTTLE VEHICLES (10,000-20,000 DAY OPERATING PERIOD) AND ON EXTRA-TERRESTRIAL BASES

In the orbital program, Project B (Figure 29) continued the development initiated under Project A, with particular emphasis on man's behavior during an extended stay in space, orbital operations (including fueling, docking and mating) and development of suitable ecological systems for Earth space stations, Lunar operations and planetary missions. The latter was a long-term program, extending from 1972 to 1992. The overall scope of this program is shown in Figures 32 and 33.

The developments accomplished in Phase B of the manned orbital Sub-Program formed the basis for Phase C as well as for the last project of Phase A of the Lunar Sub-Program (Temporary Lunar Base) Phase A of the manned planetary Sub-Program vehicle also profited from the earlier projects of Phase A of the Lunar Program. A survey of the lunar transportation systems developed for Phases A, B, and C of the manned Lunar Sub-Program is presented in Table 6.

The first manned planetary mission is Phase B of the manned planetary Sub-Program required, in addition to Phases B and A of the orbital and Lunar Sub-Programs, the operational availability of the NERVA engine and the results of Voyager flights to Mars and Venus. The latter are shown in Phase B of the unmanned planetary Sub-Program.

The Voyager program to Mars terminated in 1980, because of the subsequent manned missions to the planet in 1982 through 1986 during which period Voyager would have flown according to the original plans which did not involve consideration of manned missions prior to 1986. This led to an early start of the outer solar system Voyager which attempted its first flight to Jupiter in 1980. The design data of this probe were based on the earlier findings of the small Advanced Planetary Probe during a series of fly-by missions. The positions of the Jovian planets, shown in Figures 34 and 35, indicate that it is feasible in the years 1976 through 1979 to reach Saturn by making use of the gravitational field of Jupiter for added acceleration and reduction in transfer drive. The Advanced Planetary Probe launchings in those years made use of this opportunity and, although not all flights were successful, a wealth of new data on Saturn was obtained. Since we today know so much more about the planet, I will not go into details here. In FY 1980 the constellation of all outer planets

Table 6. Cislunar Transportation Systems 1970-1990 (Part 1 of 2)

MISSION & TRANSPORTATION SYSTEM	EARTH RETURN VEHICLE	APOLLO CAPSULE (APOLLO-1) (36,000 ft/sec)	→	→	APOLLO-2 (25,000 ft/sec)
	EARTH RETURN MODE	DIRECT ATMOSPHERIC ENTRY (DAE)	→	→	→ 11,000 ft/sec RETRO-MANEUVER
	DESTINATION SPACE VEHICLE (DSV)	LEM	LEM (ORS)[+] & CSM*	LEM (SHELTER) (U)** LEM (TAXI) (M)***	→ → LEM (SUPPLIES) (U)
	DESTINATION MISSION MODE	•LANDING •CAPTURE	CAPTURE	CAPTURE AND/OR LANDING	•CAPTURE ONLY •LANDING •CAPTURE
	CISLUNAR INTER-ORBITAL SPACE VEHICLE (CISV)	S-IV/CSM	→	→	→
	EARTH LAUNCHED PERSONNEL TRANSPORT (ELPT)	SATURN V	→	→	→
	EARTH LAUNCH VEHICLE (ELV)	SATURN V	→	SATURN V (DOUBLE LAUNCHING)	SATURN V (MULTIPLE LAUNCHING)
	FISCAL YEAR	1970	1971/72	1971/72	1973/74
	PROJECT	APOLLO LUNAR SURFACE EXCURSION SYSTEM	APOLLO LUNAR ORBITING RECONNAISSANCE STATION	APOLLO EXTENDED LUNAR SURFACE EXCURSION SYSTEM	

* CSM = COMMAND & SERVICE MODULE

** = UNMANNED

*** = MANNED

[+] ORS = ORBITAL RECONNAISSANCE STATION

Table 6. (Part 2 of 2)

Table 6. (Part 2 of 2)

MISSION & TRANSPORTATION SYSTEM	EARTH RETURN VEHICLE	APOLLO-2	RENDEZVOUS WITH ROT	→	↑
	EARTH RETURN MODE	DIR ATMOSPH ENTRY 11,000 ft/sec RETRO-MANEUVER	RETURN INTO NEAR-EARTH SATELLITE ORBIT	→	DIRECT ATMOSPH. ENTRY (CIRC. VEL. OR LESS) AND GLIDE LANDING
	DESTINATION SPACE VEHICLE (DSV)	LUNAR ASCENT & (DIRECT) RETURN STAGE	LUNAR LANDING VEHICLE (DESCENT & RETURN TO SHUTTLE)	→	↑
	DESTINATION MISS MISSION MODE	•LANDING •CAPTURE	•LANDING •CAPTURE	→	→
	CISLUNAR INTER-ORBITAL SPACE VEHICLE (CISV)	•UPRATED S-IV (LUNAR CAPT & DESC) •NUCLEAR (NERVA II) (EARTH DEPARTURE)	CISLUNAR SHUTTLE NUCLEAR (METAL REACTOR)	CISLUNAR SHUTTLE NUCLEAR ELECTRIC (MHD)	↑
	EARTH LAUNCHED PERSONNEL TRANSPORT (ELPT)	SATURN V (MOD)	REUSABLE ORBITAL TRANSPORT (ROT)	MHD-ROT ST. 1: TURBO-SCRAMJET ST. 2: NUCLEAR-MHD	↑
	EARTH LAUNCH VEHICLE (ELV)	SATURN V (MOD)	POST-SATURN	→	MHD AERO-SPACE VEHICLE
	FISCAL YEAR	1976 AND ON	1980 AND ON	1985 AND ON	1990 AND ON
	PROJECT	ADVANCED APOLLO EXTENDED LUNAR SURFACE EXCURSION SYST (TEMPORARY LUNAR BASE = PERMANENT LUNAR BASE PRECURSOR)	STATIONARY PERMANENT LUNAR SCIENTIFIC BASE AND MOBILE SCIENTIFIC LUNAR BASE	LARGE BASE UTILIZING LOCAL RESOURCES AND PLANETARY SPACE PORT	→

Table 7. Heliocentric Transportation Systems (Part 1 of 3)

MISSION & TRANSPORTATION SYSTEM	EARTH RETURN VEHICLE	BLUNT ENTRY CAPSULE ($\leq$ 42,000 ft/sec	→	→	NONE	PERSONNEL TRANSPORT	PERSONNEL TRANSPORT
	EARTH RETURN MODE	DIRECT ATMOSPHERIC ENTRY	→	→	PICK-UP BY A VEHICLE ARRIVING IN SUBSEQUENT MISS. WIND.	CAPTURE IN ELLIPTIC EARTH SATELLITE ORBIT. FROM THERE TRANSPORT TO ORBITAL HOSPITAL	RETURN INTO EARTH SATELLITE ORBIT (ORBITAL HOSPITAL)
	DESTINATION SPACE VEHICLE (DSV) AND PAYLOADS	HISV = HELIOCENTRIC INTERORBITAL SPACE VEHICLE	→	UNMANNED PROBES	• ORBITAL RECON. STA. (ORS) • 3 SURF. EXC. VEHICLES • 1 MOON EXCURSION VEHICLE		• ROBOTS • SOLAR PHYSICS STATION • MERCURY SURFACE LANDER (MHD)
	MISSION MODE	HELIOCENTRIC EARTH RETURN MANEUVER	→	• VENUS: FLY-BY • MARS: CIRCULAR CAPTURE	1-WAY TO MARS	• RETURN TO EARTH VIA VENUS • PICK-UP OF 1984 CREW • MARS CAPTURE	• RETURN TO EARTH • SURFACE LANDING • MERCURY CAPTURE
	HELIOCENTRIC INTERORBITAL SPACE VEHICLE (HISV)	• SOLAR HEAT EXCHANGER (SHE) POWERED STAGE (HELIOCENTRIC EARTH RETURN MANEUVER) • UPRATED S-IV (EARTH DEPARTURE MANEUVER)	• SHE STAGE • UPRATED S-IV (HELIOCENTRIC EARTH RETURN MANEUVER) • NUCLEAR STAGE (NERVA II) (EARTH DEPARTURE MANEUVER)	• SOLAR HEAT EXCHANGER (HELIOC. CORR. • NUCLEAR STAGE (METAL) (MA DEP.) • CHEMICAL STAGE (O_2/H_2) (MA AR.)** • NUCLEAR STAGE (NERVA II)(EA DEP.)	• STOR. CHEM. STAGE (ORBIT TRIM) • NUCLEAR STAGE (METAL) (MARS CAPTURE MANEUVER) • NUCLEAR STAGE (NERVA II) (EARTH DEPARTURE MANEUVER)	ESSENTIALLY AS 1982 MARS VEHICLE	CONTROLLED THERMO-NUCLEAR REACTOR (CTR) HISV
	EARTH LAUNCHED PERSONNEL TRANSPORT (cf. Fig. 31)	SATURN V (MOD)	ROT	ADVANCED ROT	ADVANCED ROT	MHD-ROT	MHD-ROT
	EARTH LAUNCH VEHICLE (ELV)	SATURN V (MOD)	→	POST-SATURN	POST-SATURN	POST-SATURN	POST-SATURN
FISCAL YEAR		1976	1979	1982	1984	1984	1988
PROJECT		HELIOCENTRIC EXCURSION MISSIONS. OBJECTIVES: TO EXERCISE EARTH ORBITAL DEPARTURE OPERATIONS; OPERATE FLIGHT HARDWARE, INCLUDING GUIDANCE & NAVIGATION THROUGH 3 MANEUVERS: EARTH DEPARTURE, HELIOCENTRIC, EARTH ARRIVAL; TRAIN FLIGHT CREWS FOR CRITICAL PLANETARY MISSION OPERATIONS; AND EXERCISE ACTUAL HELIOCENTRIC ABORT OPERATION.		MARS CAPTURE MISSION FOR DETAIL MAPPING AND EXPLORATION FROM ORBIT WITH ADVANCED SENSORS AND UNMANNED PROBES	ESTABLISHMENT OF AN ORBITAL RECONNAISSANCE STATION AROUND MARS FOR A NEARLY SYNODIC PERIOD. ORS[1] CARRIES PILOTED SURFACE EXCURSION VEHICLES AND MARS MOON EXCURSION VEHICLES TOGETHER WITH UNMANNED PROBES (SOME OF WHICH RETURN FROM MARS SURFACE), SCIENTIFIC LABORATORIES AND VERY ADVANCED SURVEILLANCE SENSORS	THIS IS A ROUND-TRIP MISSION IN CONTRAST TO 1984 WHICH IS A ONE-WAY MISSION. OBJECTIVE IS TO PICK UP ORS CREW AND RETURN TO EARTH	ESTABLISHMENT OF A SOLAR PHYSICS STATION ON MERCURY

* EA = EARTH
** MA = MARS

1) ORBITAL RECONNAISSANCE STATION

Table 7. (Part 2 of 3)

Table 7. (Part 2 of 3)

MISSION & TRANSPORTATION SYSTEM	EARTH RETURN VEHICLE	PERSONNEL TRANSPORT	→	→	AS 1988	→
	EARTH RETURN MODE	RETURN INTO EARTH SATELLITE ORBIT (ORBITAL HOSPITAL)	→	→	AS 1988	→
	DESTINATION SPACE VEHICLE (DSV) & PAYLOADS	DESCENT & LANDING WITH NUCLEAR PULSE HISV	→	• ROBOTS • BASE MODULES • MERCURY LANDER (MHD)	• PHOBOS RESEARCH STATION • MARS SURFACE ROBOTS • MARS LANDERS • DEIMOS EXCURSION VEHICLE	NONE
	MISSION MODE	• RETURN TO EARTH • RE-ORBITING • VENUS LANDING • VENUS CAPTURE	• RETURN TO EARTH • ESTABLISH & START AUTOMATIC SCIENTIFIC STATION • RENDEZVOUS ICARUS	AS 1988	• RETURN TO EARTH • LANDING ON PHOBOS • MARS CAPTURE	• RETURN TO EARTH • FLY THROUGH HEAD OF COMET ENCKE
	HELIOCENTRIC INTERORBITAL SPACE VEHICLE (HISV)	NUCLEAR PULSE (NP) HISV	→	CONTROLLED THERMONUCLEAR REACTOR (CTR) HISV	CTR HISV	NP HISV
	EARTH LAUNCHED PERSONNEL TRANSPORT	MHD-ROT	→	→	MHD-ROT	→
	EARTH LAUNCH VEHICLE (ELV) (Fig. 31)	POST-SATURN	→	→	POST-SATURN	→
FISCAL YEAR		1988	1989	1990	1990	1991
PROJECT		MANNED LANDING ON SURFACE OF VENUS BY MEANS OF A NUCLEAR PULSE VEHICLE AND LIMITED EXPLORATION OF SURFACE	PLANTING AN AUTOMATIC SCIENTIFIC STATION ON ASTEROID ICARUS WHOSE PERIHELION LIES INSIDE MERCURY ORBIT AND APHELION BEYOND MARS ORBIT	FIRST OF A SERIES OF MERCURY SURFACE EXPLORATION MISSIONS	ESTABLISHMENT OF A RESEARCH AND SUPPLY (FOR LANDING PARTIES) STATION ON MARS MOON PHOBOS WHICH IS WELL ACCESSIBLE FOR LOW-THRUST VEHICLES	EXPLORATION FLIGHT OF A MANNED NUCLEAR PULSE HISV THROUGH THE HEAT OF COMET ENCKE TO STUDY ENVIRONMENT IN GREATER DETAIL THAN WAS DONE FORMERLY WITH PROBES

Table 7. (Part 3 of 3)

MISSION (FY)	HEM (1976)	HEM (1979)	MARS CAPTURE VENUS FLYBY (1982)	MARS ORBITAL RECONNAISSANCE STATION & SURFACE EXCURSION (1984)	
				PARTY NO. 1	PARTY NO. 2
DEPARTURE EARTH	4 JULY 1976	16 NOVEMBER 1978	2 JANUARY 1982	20 MARCH 1984	21 MARCH 1985
EARTH DEPARTURE MANEUVER (ft/sec)	11,000	11,700	12,500	14,100	14,100
OUTBOUND TRANSFER TIME (d)	20	30	200 (Ea-Ma)	220 (Ea-Ma)	171 (Ea-Ve) 184 (Ve-Ma)
HELIOCENTRIC ABORT MANEUVER (ft/sec)	7,600	13,500	-	-	-
MARS CAPTURE MANEUVER (ft/sec)	-	-	14,000	10,000	19,200
ARRIVE MARS	-	-	21 JULY 1982	27 OCT. 1984	11 MARCH 1986
MARS CAPTURE PERIOD (d)	-	-	69	529	29
DEPART MARS	-	-	28 SEPT. 1982	9 APRIL 1986	
MARS DEPARTURE MANEUVER (ft/sec)	-	-	14,000	9,000	
RETURN TRANSFER TIME (d)	60	100	149 (Ma-Ve) 161 (Ve-Ea)	176 (Ma-Ea)	
ARRIVE EARTH	22 SEPT. 1976	26 MARCH 1979	4 AUGUST 1983	2 OCTOBER 1986	
ATMOSPHERIC ENTRY VELOCITY (ft/sec)	36,400	36,600	39,000	CAPTURE IN EARTH SATELLITE ORBIT MANEUVER: 10,700 ft/sec	
OVERALL MISSION PERIOD (d)	80	130	579	925	560
OVERALL MISSION VELOCITY NOT COUNTING SPECIAL MANEUVERS AT TARGET PLANET (ft/sec)	18,600	25,200	40,500	24,100	53,000

Ve = VENUS; Ea = EARTH; Ma = MARS

NOTE: ALL IMPULSE VELOCITIES, EXCEPT THOSE OF HELIOCENTRIC ABORT MANEUVER, ARE MULTIPLIED BY 1.04 TO ACCOUNT FOR LOSSES.

ARE MULTIPLIED BY 1.04 TO ACCOUNT FOR LOSSES.

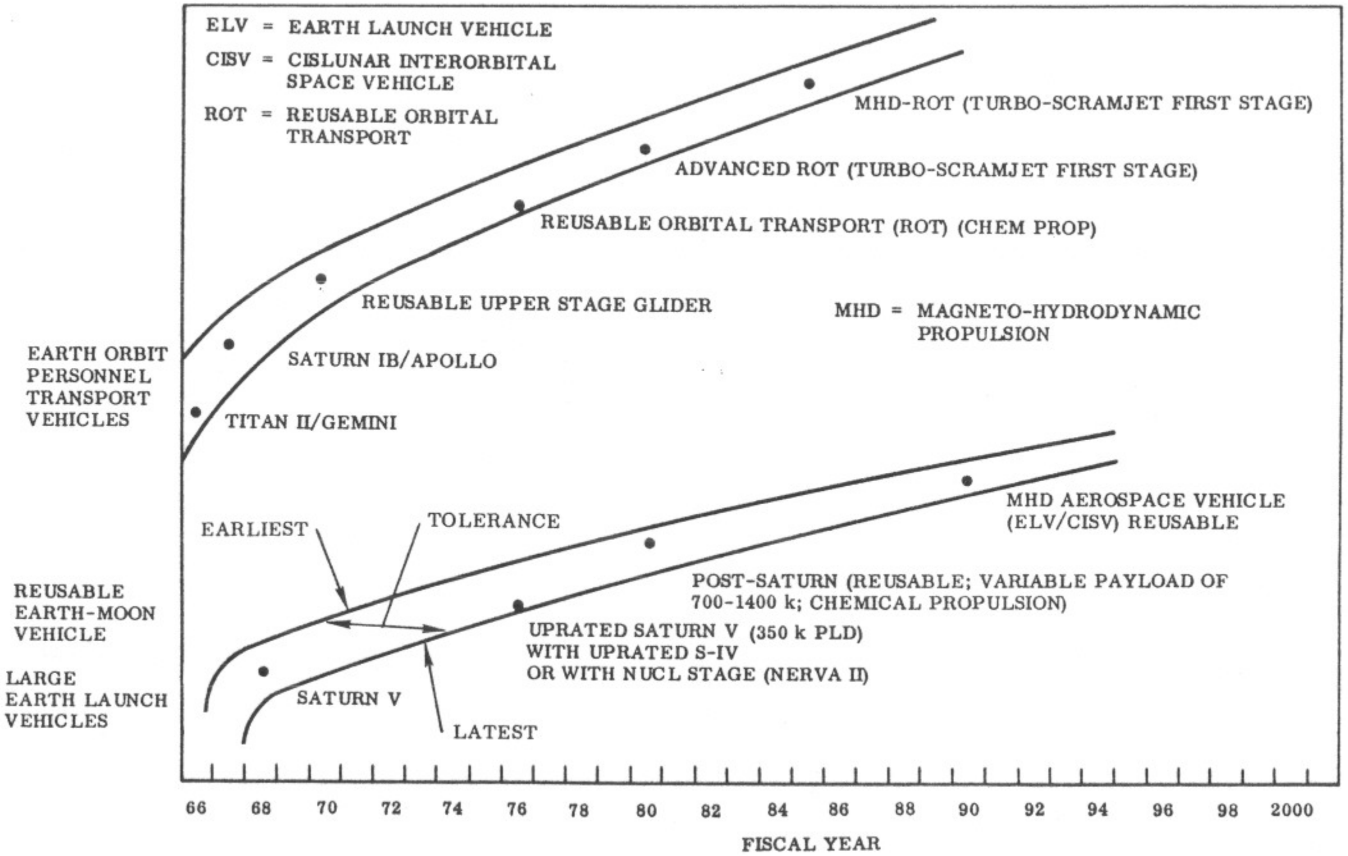


Figure 31. Earth Launch Vehicle Evolution

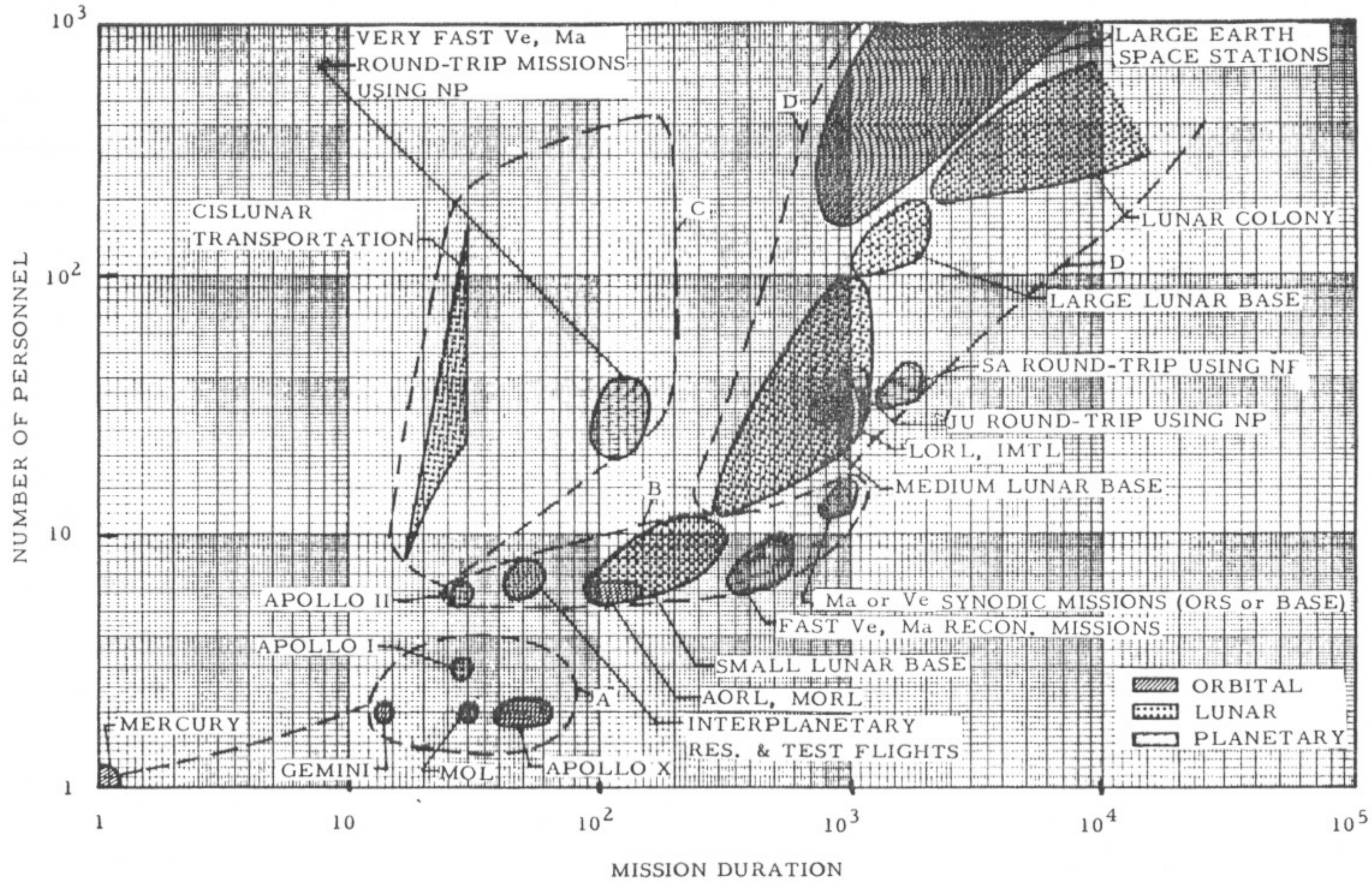


Figure 32. Mission-Oriented Evolution of Ecological Systems

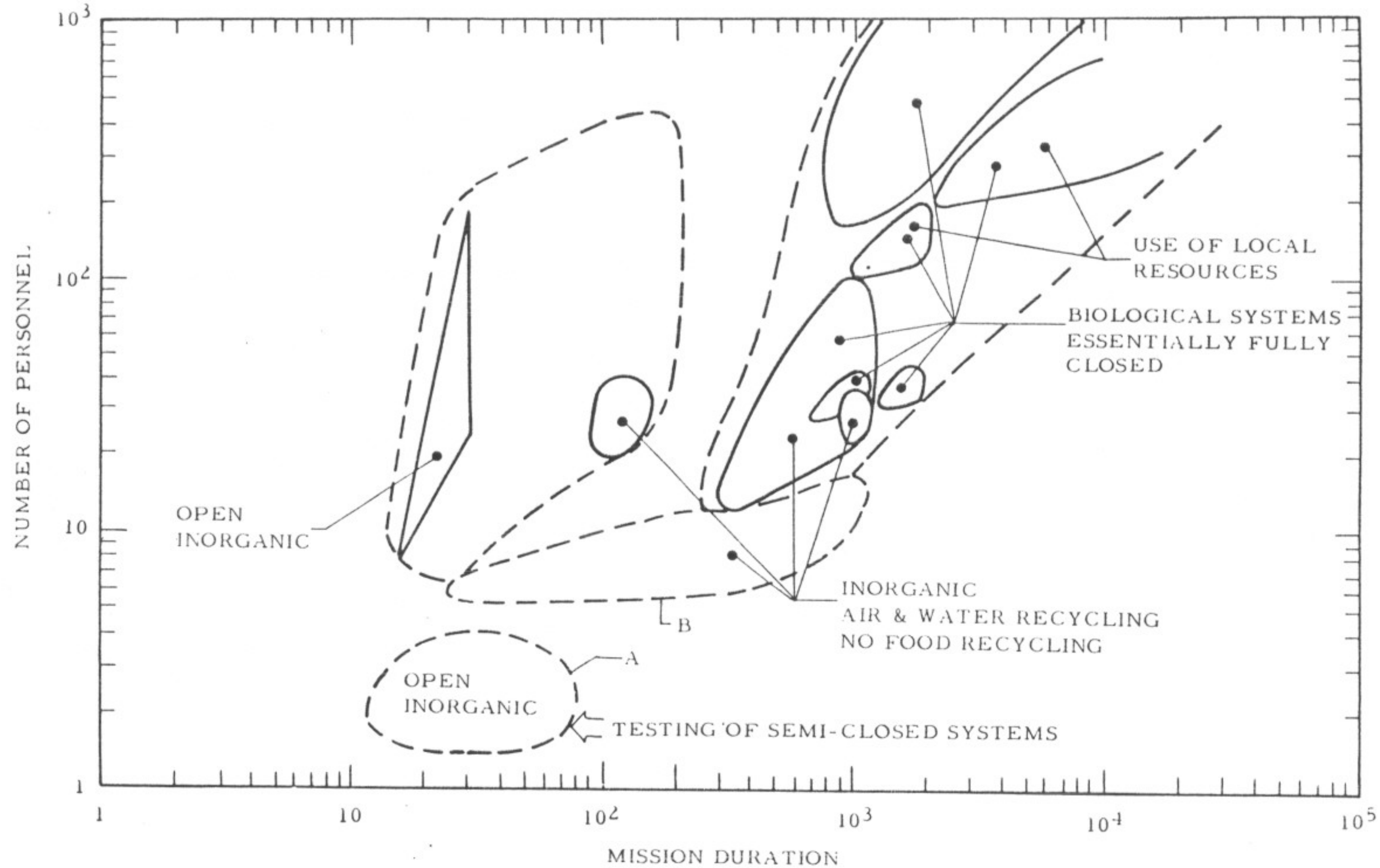


Figure 33. Correlation of Type and Capacity of Ecological Systems

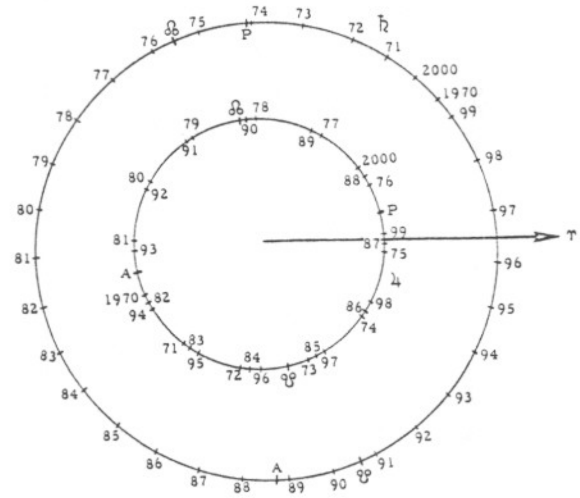


Figure 34. Positions of Jupiter and Saturn 1970-2000. Positions Refer to the Beginning of the Year Indicated (First Days of January)

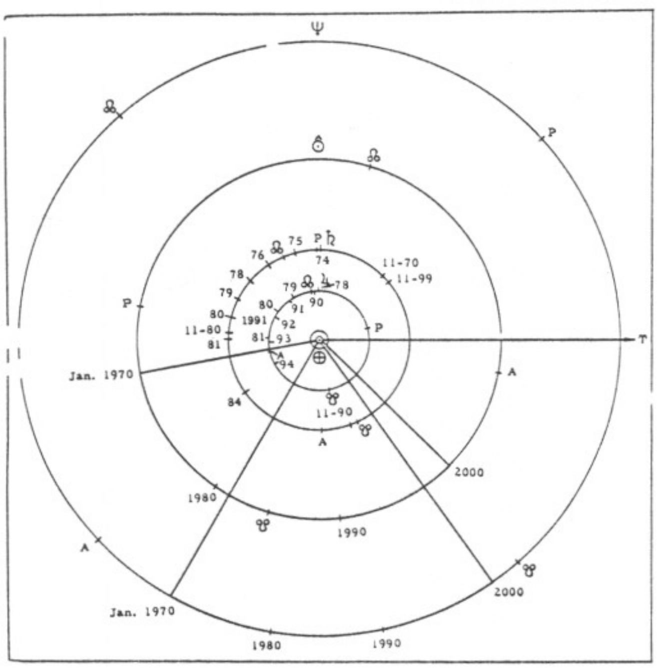


Figure 35. Positions of Jovian Planets 1970-1990. Positions Refer to the Beginning of the Year Indicated (First Days of January)

permitted theoretically their use as gravitational acceleration as one after the other was bypassed by the probe. An attempt was made to take advantage of this opportunity, but the mission was not successful. The error sensitivity of a hyperbolic encounter, together with the limited navigational maneuvering capability of the small Advanced Planetary Probe caused the probe to be thrown off course (in direction and path inclination) during the Saturn encounter. The probe passed Uranus at thirty times the planned distance and missed Neptune altogether. The USSR who tried the same mission, was on course at least up to Uranus, to judge by the osculating heliocentric orbit element, following Saturn fly-by but they received no further data shortly after the probe left the activity sphere of Saturn. As you know, very interesting results were obtained with the comet fly-through probe missions to D'Arrest and Kopff, but the third one failed to yield any data.

While the frontier of the unmanned probe program moved into the outer solar-system, Phase B of the manned planetary sub-program was under way. The Mars capture mission in 1982 was a complete success (Figure 36). The desirability of a subsequent surface excursion, but not of a synodic base, of some 250 days was indicated. The decision was made to fly out again in 1984, but on a one-way mission, so that the same vehicles as in 1982 could be used but with more than three times the payload (about 400,000 pounds). This payload consisted of a more elaborate orbital reconnaissance station, three surface excursion vehicles (chemically powered), and one Mars-Moon excursion vehicle, (nuclear-electrically powered).

Table 7 surveys che heliocentric transportation systems which were deployed in the 1976 through 2001 time period.

In 1988 NASA put the first NP and CTR vehicles into service.

A convoy of two NP vehicles conducted its maiden voyage to planet Venus. One vehicle was aerodynamically streamlined and heavily protected for descent to the surface of the planet. The convoy transferred at a brisk pace and reached Venus 60 days later. Some of the crew members who had participated in the earlier Mars expeditions stated the difference was like changing from a slow, old fashioned freighter to a modern ocean

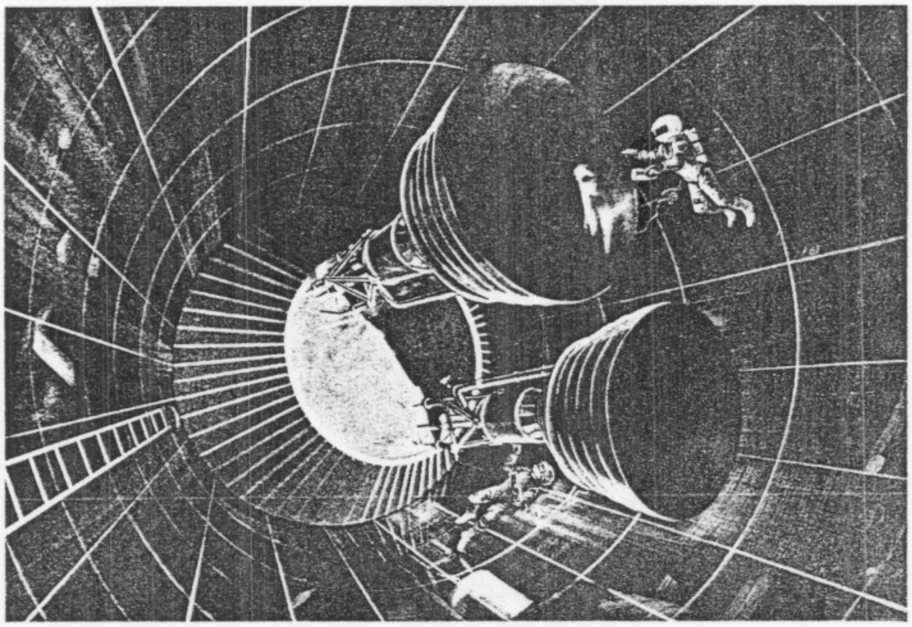


Figure 36. Mars Capture Mission in 1982. Orbit Crew Inspects the Nuclear Twin Engine NERVA II System of the Earth Departure Module. Each Engine Delivers 250,000 lb of Thrust.

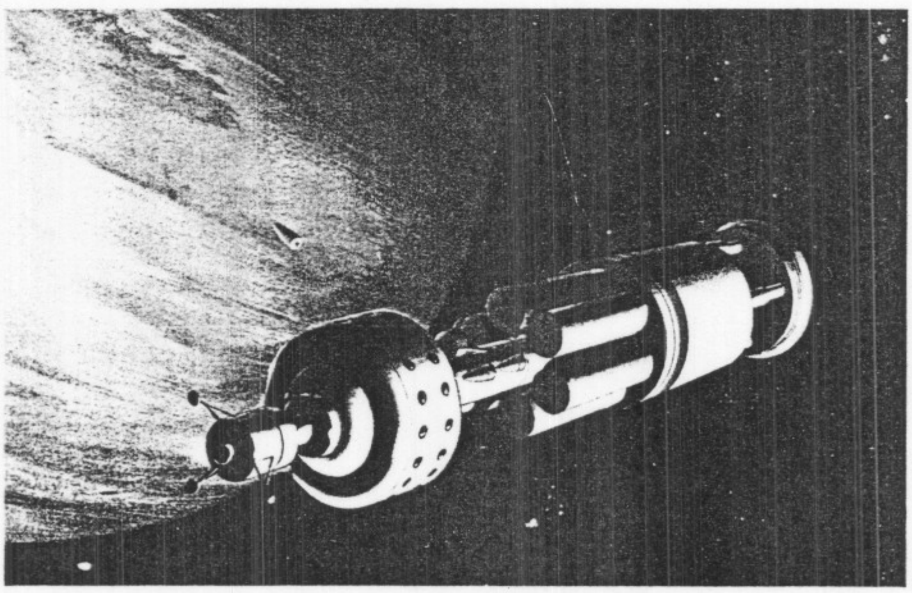


Figure 37. Nuclear Pulse Convoy at Venus, 1988. In Foreground the Inter-Orbital Nuclear Pulse Vehicle. In Background, Descending Toward the Venus Cloud Cover, the Nuclear Pulse Surface Excursion Vehicle. At Lower Left the Newly Discovered Tiny Venus Moon Cupid I

liner. Soon following capture-orbit establishment, the landing vehicle broke away and fell toward the cloud cover of Venus (Figure 37). The orbiting vehicle discovered a tiny moon of Venus, about 1-1/4 miles in diameter. They called it Cupid I, just in case there were more little Cupids around. The descent vehicle stayed for two weeks on the surface of Venus and thereafter staged a successful re-ascent through the atmosphere. The pictures taken by the flight crew, showing the awesome glow of the nuclear charges in the darkness of the dense Venus atmosphere, are among the most impressive ever taken during any helionautical mission. The exciting story of their surface excursion and the unexpected discoveries made by them are history; but more missions to the Venus surface will be made in the future.

The maiden voyage of the CTR to Mercury was equally successful (Figure 38) and the double-headed third phase (Phases C and D of the manned planetary sub-program) was off to a good start. A short time after their landing in the polar twilight zone of Mercury, the landing party, exploring their surroundings, discovered fabulously wide gold-silver-lead veins, cobalt veins, and vanadium veins. The thickness of one gold vein, in ore with lead, was 30 feet across by 80 feet in depth. Two cobalt veins were more than 100 feet across and 70 feet in depth.

These findings resulted in a number of follow-on flights to Mercury for surface exploration during the years of 1990 to 1993. On Earth, the economic feasibility of such finds would have been sensational. On Mercury, the relative ease of mining, coupled with unlimited power from nuclear reactors as well as from the Sun and the fact that a new super-CTR of 500 tons cargo capability from Mercury to Earth would be put into service shortly made the exploration of these veins economically attractive, provided the ore was processed on the spot, to maximize the payload yield. Under those conditions, the transportation cost in the new CTR freighter will soon be of the order of $225 per pound of refined metal.

In the meantime the NP vehicles have carried out numerous exploration missions, as indicated in Figure 30. These included a flight through the head of comet Encke, exploration missions to Jupiter (Figure 39) including

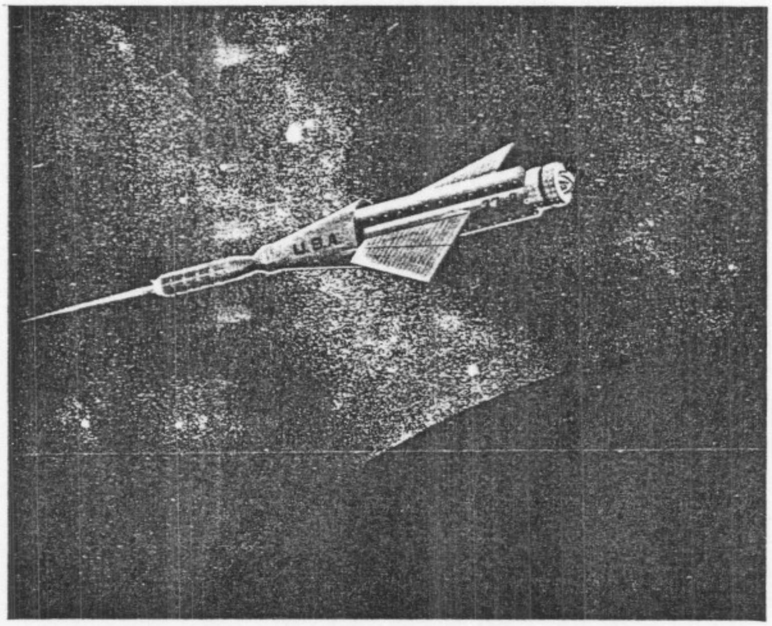


Figure 38. CTR Passenger Carrier 37-P Over Midnight Point of Mercury, 1988, In Exciting Capture Maneuver for Subsequent Descent and Landing of Toroidal Module at Front End of Vehicle, Representing the Solar Physics Research Station. A Team of Six Scientists and Six Engineers Descended with the Module for a Surface Stay Time of 9 Months.

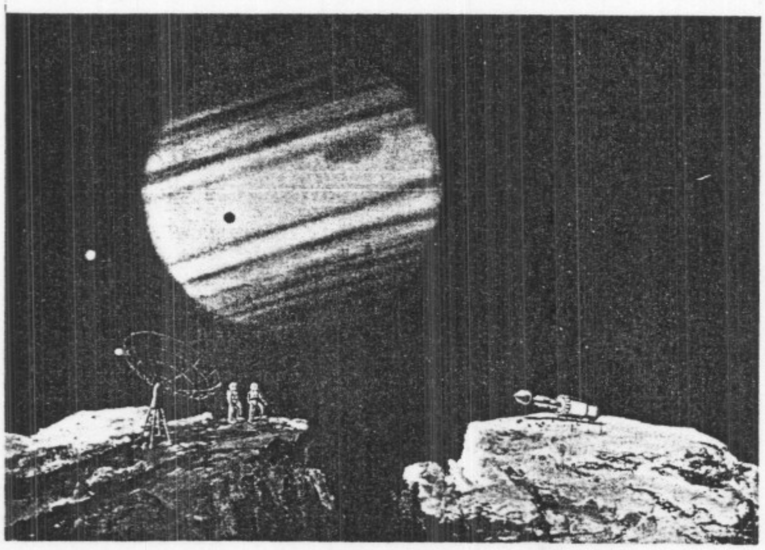


Figure 39. Explorer Party Lands on Jupiter VII, 1997

the establishment of a research base on Callisto and a test drive into the outer atmosphere of the planet, a feat which can be accomplished only with a nuclear pulse system.

By now, the constellation has recurred which permits rapid flights to Saturn with the help of Jupiter's gravitational field. We have taken advantage of this fact by undertaking a number of joint Jupiter-Saturn missions in which a convoy of Jupiter- and Saturn-bound ships leaves for Jupiter. At Jupiter approach, the convoy separates. The Saturn vehicles go through a gravitational encounter mode with Jupiter and continue on the way to their target. The Jupiter ships enter into an elliptical-orbit capture mode and subsequently orbit-maneuver to the moon Callisto.

The joint Jupiter-Saturn mission series was initiated in 1996 (Figure 30). Based on the findings of the unmanned probes to Jupiter in the 70's through 90's (Figure 30), sufficient information was available to decide right away to establish a scientific base on Callisto, the outermost of Jupiter's Galilean moons. Since then, Callisto has become the main base in the Jovian system. Long-term observations of the planet, and probing of its atmosphere to varying depths by radar, by unmanned probes and by manned vehicles have been initiated. From Callisto, visits have been paid to almost all other moons of the Jovian satellite system and various self-sufficient instrument packages, in fact, entire automatic research stations, have been placed on them. They report back to Callisto and are maintained from the manned Callisto base.

In contrast to Jupiter, the first manned mission to Saturn was exploratory, but it included an excursion to the surface of Titan, Saturn's largest moon. Two years after the first mission, a convoy left for Saturn with the goal of establishing a manned scientific base on Titan. The objectives of this base relative to Saturn and the Saturnian satellite system are, as you know, analogous to those of the Callisto base. We are not as far along with Saturn as we are with Jupiter, but as the FY 2001 wears on, we are getting settled for a first look around in Saturn's realm.

The Outlook in FY 2001

I must apologize for having looked back all this time. The space age individual looks forward, more so and perhaps more confidently in view of the trials passed and the tasks accomplished, than any generation before him in the history of mankind. Therefore, as we stand on the threshold of a new millenium, I hope you will permit me a brief look at where we might go from here.

First of all, the more mundane tasks. We can expect, in the next 50 years, a rapid expansion of raw material exploitation, as bigger and less expensive transportation systems become available. This exploitation will include prospecting through asteroid hunting by us and by the Russians and Chinese; as well as expansion of our facilities on Mercury. Incidentally, we are getting company there too, judging by the new Russian 30-year plan. I am happy to say that, under the United Nations General Assembly Resolution of March 1990, we need not foresee any trouble with others sharing in the Mercurian raw material wealth. This resolution, as you will remember, provides that "rightful occupation of extra-terrestrial territory is contingent upon, and limited to, the capability of developing this territory", i.e. base establishment or metal ore mining. Just to land, drop flags or pellets or other symbols is merely "evidence of visitation" not "rightful occupation". By establishing a base, for instance, you occupy a certain area, legally defined by size and purpose of the base, rather than the entire celestial body. Exceptions are bodies of less than 10 miles in diameter.

Solar system fertilization is, so far, restricted to Mars and even there it is only in its initial stage. As you know, some very bold plans to modify the atmosphere of Venus have been submitted which are presently being examined by the special UN Astro-Fertilization Committee. The beforementioned UN Resolution of 1990 does not apply to fertilization projects wherever these are of global scale. In such instances, the project is of international scope with voluntary participation under UN supervision and control.

The solar transportation picture will brighten substantially within the next 20 years. With the β-value of the CTR vehicles now exceeding 0.9, we have the first two helionautical transporters of 5000 ton payload capability under construction. Moreover, both the United States and Russia have plans completed for greatly improved NP vehicles for outer solar system transportation using large nuclear fusion pulse units of megaton size. What makes these new systems so economic is not only their size but virtually complete independence from terrestrial propellant supply. They can use light or heavy extra-terrestrial material. Due to their size, the pulse units can be detonated, and any type of propellant be "plasmatized", at considerably greater distance from the pusher plate than in the earlier models. By means of an "electromagnetic lense", the plasma is then focused magnetohydrodynamically on the pusher plate, causing a uniform plasma jet to hit the pusher plate, resulting in better propellant utilization and in higher specific impulse than in the earlier models. If applied to inner solar system transportation these new vehicles will have a payload capability in excess of 20,000 tons, reducing the transportation cost of metal from Mercury to $2/lb or less.

Thus, for personnel and special equipment transports we will, in the future, use the new CTR systems; even for flights to our bases in the outer solar system to which the CTR ships will travel through "cleaner" space regions along extra-ecliptic flight paths. For large cargo loads either way, we will use the new NP vehicles. For further exploration of the atmospheres of the outer planets we have large probes under construction capable of floating for many years in operative condition at great depths in the atmospheres of these planets. For the next 25 years we won't be able to use Jupiter's g-field for flights to Saturn; but with our new vehicles we no longer are so dependent upon such assistance, although we certainly will use it whenever it is available; anything to improve our cost effectiveness. Thus, we will continue to build up our Jovian and Saturnian activities and, before the new decade is over, go beyond, to Uranus, Neptune and Pluto.

But the most exciting outlook is in the energy conversion area which, of course, is the basis of all propulsion, namely, in the field of anti-matter. As you know, anti-matter has been produced in tiny quantities already 35 years ago. Increasing the production rate was quite a problem; but even more formidable, indeed, seemingly unsolvable, was the problem of storing anti-matter and releasing it in a controlled manner for the purpose of energy generation. We now know that magnetic field storage of anti-matter is feasible and the big race is on among all leading powers to be the first to achieve practical utilization of these concepts. In this connection, it is appropriate to mention the intense preoccupation of the Chinese helionautic activities with the change and control of asteroid orbits. This, of course, does potentially have significant practical consequences. Last year, the Chinese succeeded in braking one of the smallest Trojan asteroids out of its orbital confinement about libration point L_5 and throwing it into an independent heliocentric orbit. This caused Senator Smarth of Wisconsin recently to call for renewed efforts on the part of the United States in pressing for a United Nations Resolution regulating the orbit changing of asteroids, especially so far as proper notification of all space-going nations, as to intended and actually accomplished orbit changes is concerned. The Chinese feat may have been accomplished by using large conventional fusion charges. But it is just possible that they were testing an anti-matter pulse unit.

I, for one, have little doubt that a workable anti-matter propulsion unit will come into existence during the next 20 to 30 years, which is suitable for driving very large, that is planetoidal, space complexes. This will give us not only the possibility of changing asteroid orbits and stabilizing the outer satellite system of Jupiter; this drive will also permit us to follow our unmanned probes into interstellar space, to Proxima Centauri, to Sirius and possibly to Prokyon. But now I am talking about galactic rather than solar transportation and this subject should be left to another symposium.

Postscript

A "look into the future" is always challenging and refreshing, so long as one does not take the predictive aspects and details too seriously.

It has been attempted to suggest here how the manned planetary program might evolve, what the state-of-the-art might be in FY 2001 and why. The two principal propulsion systems chosen have at least in their favor the logic of a specific impulse which is commensurate with the energy requirements of solar system flight in not unreasonable travel times; the logic of growth potential in specific impulse (NP) and of thrust acceleration (CTR); the logic of greater simplicity than alternate propulsion systems; and in the case of the NP also the logic of its capability to enter into and emerge from extremely unfriendly planetary atmospheres; at least better than any alternate device.

Of course, this does not alter the fact that these drives are not needed if there is no requirement for missions beyond Venus and Mars, or for a particularly economical approach to even the extended missions; for both are extreme systems, expensive to develop and, at least the NP, politically controversial at this time. That, however, does not change their eminent technological suitability as means of solar transportation.

The question of whether missions such as those described here will be undertaken during the 1988-2001 period, depends at least as much on the world situation as on the technological state-of-the-art. Since it is obvious that a world war or even a critical number of smaller wars will severely hinder man's progress in astronautical exploration as well as in numerous other areas, this case is really trivial and not of particular interest in a meeting which obviously assumes that there will be a space age in FY 2001. Therefore, one really has not much choice but to assume a sane world which allows the space age to flourish. After all, one can at least hope that this will be an accurate assumption. At least for a short while it may serve a good purpose to risk being naive by assuming that agreements will be fair, hence honorable; that the excitement of space exploration will still have universal appeal; that we like to follow greater goals and that massive deployment of nuclear pulse units will cause nuclear bombs to evaporate in the wake of solar transportation systems. It may be a naive notion, but at the very minimum it is a pleasant thought.

ACKNOWLEDGEMENT

Figures 1, 2, 4, 5, 36, 38 and 39 have been reproduced with the kind permission of General Dynamics/Convair from the GD/C-produced motion picture, A Study of Manned Interplanetary Missions (Mrs. Betty Miller, Director).

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