SCIENCE APPLICATIONS, INC.

Report No. SAI 1-120-839-S4

MULTIPLE DISCIPLINE
SCIENCE ASSESSMENT

by

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Contract No. NASW-3035

December 1978

FOREWORD

This technical report is the final documentation for Task 4: Multiple Discipline Science Assessment of NASA Contract No. NASW-3035 titled "Advanced Planetary Studies." This study was performed between April 1977 and January 1978 by the Space Sciences Division of Science Applications, Inc. for Lunar and Planetary Programs (Code SL), Office of Space Science, NASA Headquarters. The results are intended to be a basis for including other science disciplines into the planning of future planetary missions.

ACKNOWLEDGEMENTS

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SUMMARY

The purpose of this report is to examine other science disciplines and to determine where and when it is appropriate to include their objectives in the planning of planetary missions. The science disciplines considered are solar astronomy, stellar and galactic astronomy, solar physics, cosmology and gravitational physics, the geosciences and the applied sciences. For each discipline, science objectives are identified which could provide a multiple discipline opportunity utilizing either a single spacecraft or two spacecraft delivered by a single launch vehicle. Opportunities using a common engineering system are also considered. The most promising opportunities identified during this study are listed in Table 1 and described briefly below.

Mercury Orbiter

A spacecraft in orbit about Mercury can acquire unique data on the Sun and on relativistic gravitational effects. The advantages for solar observations are a five to ten times greater solar flux, a longer time for observing individual features and the possibility of using Mercury as an occulting disk for coronal studies. From a much more accurate determination of Mercury's orbit, information on the internal structure of the Sun can be derived and tests can be made on relativistic gravity theories.

To accomplish any or all of the above objectives there must be changes in the set of instruments, in the spacecraft systems or in the spacecraft operations. For example, any useful solar observations will require several instruments with high spatial and/or spectral resolution to investigate the disk and the corona at visible and ultraviolet wavelengths. Another desirable instrument is a neutron detector, since the flux of solar neutrons is greatly attenuated at the Earth by radioactive decay. To determine Mercury's orbit accurately either the spacecraft must be "quiet," i.e., at least for several orbits on a regular

Table 1

PROMISING MULTIPLE DISCIPLINE SCIENCE OPPORTUNITIES

Type of Commonality	Planetary Use	Additional Disciplines	Relevant Observations
Single spacecraft	Mercury Orbiter	Solar astronomy; Gravity physics	Solar images; Relativistic effects
	Mars Sample Return	Solar physics; Applied science	Collection of samples exposed to solar particles
	Neptune or Pluto Flyby	Solar physics; Stellar astronomy	Interstellar neutral H and He; Magnetic field, cosmic rays
	Any Mission	Solar physics; Stellar astronomy	Fields, particles; Gamma ray bursts
Single launch vehicle	Mercury Orbiter	Solar astronomy; Solar physics	Solar images from 0.2 AU synchronous orbit
	Mars Orbiter	Solar astronomy; Solar physics	Solar images and particles from 90° orbit
	Neptune or Pluto Flyby	Solar astronomy; Solar physics	Solar data down to 0.02 AU or from 90° orbit
Single system	Atmospheric Probe	Geoscience	Upper atmosphere structure, composition
	Remotely Piloted Vehicles	Geoscience	Atmosphere structure, composition

basis, there must be no unknown forces acting upon it. Some improvement in spacecraft tracking procedures and equipment may be needed.

The additional mass for science instruments is estimated to be between 30 and 90 kg and that for spacecraft subsystems other than propulsion is taken as 15 to 45 kg. If the orbiter is in a circular polar orbit, 25 to 75 kg of additional propellant is required to put the extra mass into orbit. For an elliptical orbit, the added propellant is only 5 to 15 kg. The resulting spacecraft has relatively complex pointing requirements since it must point instruments at the Sun and Mercury, communicate with the Earth and control its temperature.

Mercury orbiter missions are delivered by low-thrust propulsion systems for which an increase in the required net mass on approach can be achieved with a longer flight time. Typically this sensitivity is 0.3 days per kilogram although a specific case could be up to a factor of two different. Thus, the additional flight time for 30 to 90 kg of additional multiple discipline science is 15 to 45 days in the elliptical orbit case and 21 to 63 days if the spacecraft is in a circular orbit.

Mars Surface Sample Return

The multiple discipline opportunity that can be easily combined with a Mars Surface Sample Return (MSSR) mission is the return of samples exposed to the deep space environment. These samples would be carefully selected and prepared so that they could be used for studies of solar wind ions, solar flare particles and micrometeoroids, and for investigations of the effects of deep space environments on materials. Analyses would be done using the many powerful techniques available in Earth-based laboratories. The samples would be deployed while in deep space and retrieved prior to capture in Earth orbit. This experiment should be part of an Earth return vehicle (ERV) that stays in orbit about Mars.

The nominal experiment returns 5 kg of samples and adds about 20 kg to the net mass of the ERV at launch. It is recommended that an additional 20 kg be allocated to a package of particles and field instruments to measure the interplanetary environment to which the samples are exposed. As an example, the overall increase in the injected mass of the ERV vehicle for a 1988 MSSR mission is about 325 kg for the nominal sample. Each additional kilogram of sample requires about 20 kg of injected mass. This opportunity is generally easily accommodated within dual launch concepts for MSSR for which the ERV usually has more than a 325 kg margin.

Neptune or Pluto Flyby

A spacecraft on a Neptune or Pluto Flyby mission offers an excellent opportunity for study of the interstellar medium and its interaction with the solar wind. These objectives require observations during the cruise phases of the mission, particularly after the planet encounter. It is presumed that most of the necessary instrumentation is already included for the purpose of measuring the interplanetary particles and fields. Some increases may be needed in the sensitivity and/or the energy ranges of these instruments. One obvious additional experiment is a detector for neutral atoms and molecules. Its impact on the spacecraft is negligible. The inclusion of this objective implies that the mission duration should be as long as possible--limited only by spacecraft reliability or communication capability. Some subsystem modifications may be advised.

Jupiter swingby trajectories are the best choice--the relevant opportunities are 1990, 1991, 1992 and 1994 for Neptune and 1989, 1990 and 1991 for Pluto. Flight time can also be decreased by using a larger launch vehicle, e.g., a Tug instead of an IUS, but this is not always advantageous because the faster trajectory has an unallowable swingby distance below Jupiter's surface. In addition, some restrictions

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on the planetary encounter may be necessary to put the spacecraft on
a post-encounter trajectory that escapes the solar system at a rapid
rate. Obviously, the fastest trajectory to Neptune or Pluto is desired.

Mercury Orbiter and Synchronous Solar Observatory Missions

A single low-thrust propulsion system can deliver an orbiter to
Mercury and a solar observatory to a 0.20 AU circular orbit. A space-
craft in a circular orbit at 0.20 AU has an orbit period of about
30 days which is also the rotation period of the solar photosphere.
Thus, this spacecraft can observe continuously any feature on the solar
disk or in the solar corona. To accomplish this objective requires us-
ing instruments with high spatial and/or spectral resolution and wide
spectral range. There are also opportunities as the Earth-Sun-spacecraft
angle constantly changes for stereo observations of the Sun. This orbit
puts severe stress on the thermal control subsystem, but there is no
reason to believe this problem cannot be solved. It is expected that
the science instrument package would be 100 to 200 kg and that the
spacecraft would be 600 to 800 kg. The latter may be reduced some if
the low-thrust system can be retained to provide power and some attitude
control functions.

These combined missions can be performed by the 60 kw Ion Drive
low-thrust propulsion system studied for Comet Halley Rendezvous but
without a concentrator. While the flight times are longer than for
either mission alone, it is possible to do many combinations. For ex-
ample, the single circular orbiter for Mercury (1200 kg) and a 600 kg
solar observatory requires flight times of 500 and 420 days, respect-
ively, when an IUS(Twin) is used, and only 775 and 690 days when the
Tug(R)/EE-Kick is used. Alternatively, the Tug(R)/EE-Kick allows the
1200 kg and 500 day Mercury mission to be performed in conjunction with
a 950 kg and 830 day solar observatory. Useful payloads can be delivered
using a somewhat less powerful and less advanced low-thrust system, but
the flight times are longer.

Missions to Planets and a Solar Polar Observatory

The out-of-the-ecliptic missions considered here result in highly inclined (>50°) circular orbits. The purpose of such a mission is to study the structure of the Sun and of interplanetary space as a function of solar latitude. There are also some possibilities for stereoscopic solar imagery and for low background astronomical observations. This solar polar observatory would have high spatial and/or spectral resolution instruments covering most, if not all, spectral regions. Total spacecraft mass is expected to be about 800 kg, including 200 kg of science. The planetary missions considered are a Mars Orbiter and Jupiter Swingby missions to Neptune or Pluto. The Mars Orbiter could be a geophysical orbiter, perhaps including penetrators or supporting some surface system. The Mars approach mass for these options would be 1200 to 1600 kg. The Neptune mission is assumed to include an atmospheric probe for a total mass of 800 kg while the Pluto case employs only the 600 kg flyby spacecraft. Swingby opportunities to Neptune and Pluto begin in 1990 and 1989 respectively, as cited above.

After injection to Earth escape by an IUS(Twin) and delivery of a 1600 kg Mars Orbiter, Ion Drive (60 kw) can then take an 800 kg payload to an inclination of 53° (the orbit period is 1.88 years). The overall flight time is about 1000 days. Reducing the Mars Orbiter to 1200 kg results in an inclination of 62° at about 1250 days. The Jupiter swingby mode can easily give a 90° inclination. Using a Tug(R)/EE-Kick for injection to Earth escape gives a solar observatory mass of 640 and 740 kg for the Neptune and Pluto missions respectively. These payloads could be increased by increasing the flight time beyond the 1280 days considered here or by going to a circular orbit larger than 1.0 AU. It may be possible to do these missions with a smaller and less advanced low-thrust system, but a longer flight time is needed to increase inclination after Mars encounter or to provide adequate payload at Jupiter via a SEEGA trajectory.

Missions to Planets and a Solar Probe Mission

Another interesting mission for investigations of the Sun is the Solar Probe mission which goes to a perihelion of 0.02 AU. This can only be done using a Jupiter swingby where there is again the opportunity to send a second spacecraft on to Neptune or Pluto. The choice of perihelion allows _in situ_ study of the solar corona at 4 solar radii and offers reasonable hope for a technical solution to the thermal control problem. Significant information is also obtained on the solar gravitational potential and effects predicted by relativistic gravitational theories. Both remote sensing optical instruments and _in situ_ particle and field instruments are desired. The range of science payloads is assumed to be 50 to 100 kg resulting in a spacecraft mass of approximately 600 to 800 kg. About 25% of this is the mass of the heat shield used for thermal control.

Both ballistic and low-thrust trajectories can be considered for these opportunities. The nominal ballistic missions are easily done by the Tug(E)/EE-Kick with flight times of about 2.4 and 7.2 to the Sun and either Neptune or Pluto, respectively. This Tug vehicle can do both missions and deliver more payload in less time than can be done for any single target by a single IUS vehicle. The orbit period of the solar probe is 5 years, typically, so the mission may be limited to only one solar encounter. Ion Drive (60 kw) can be used after the Jupiter swingby to reduce the period of this orbit to between 1.2 and 2.0 years depending on the spacecraft mass and the launch vehicle.

Atmospheric Probe

Atmospheric probes have been used or are planned for planetary studies at Venus, Jupiter, Saturn and Titan. In all cases, the major objective is to obtain a vertical profile of basic _in situ_ data on the structure and composition of the atmospheres. The average properties of the Earth's atmosphere are well known. Variations are studied using

aircraft, balloons and sounding rockets. In the future, this detailed vertical structure information for both the Earth and the planets could be obtained with atmospheric probes. The proposed concept for using probes at Earth is based upon delivery of many probes to orbit as a partial Shuttle payload and recovery of all systems for subsequent reuse. This is necessary to make this approach cost competitive with other ways to obtain similar data. If such a concept can be developed, then a new technology base is established which reduces the design and construction expenses for subsequent, more sophisticated planetary probes.

Remotely Piloted Vehicles

A remotely piloted vehicle (RPV) could be used in the exploration of a planet with an atmosphere, particularly Venus and Mars. The air-breathing RPV is now a well-developed concept for both military and civilian applications on the Earth. The preferred system would be designed to operate at a pressure of 5 mbar and can be used at an altitude of about 40 km on the Earth or near the surface of Mars. At Mars, the RPV could be used to study the atmosphere, obtaining horizontal profiles of its properties, to look at the surface with high spatial resolution remote sensing instruments or to transport small payloads, such as surface samples or small experiment packages. Conceptual designs of airplanes to operate in thin atmospheres for long durations are characterized by high lift-to-drag ratios and large dimensions-- the same characteristics found in gliders. For common applications, the source of power must be able to operate in a CO_2 atmosphere (e.g., a hydrazene engine, a primary battery or a nuclear thermal generator).

Fields, Particles and Gamma Ray Bursts

Particles and field observations have frequently been included in the scientific payloads of planetary missions. There is a continuing need for particle and field data at heliocentric positions other than

that occupied by the Earth. Particle data are desired for the solar wind ions and electrons, the solar flare particles and the low energy cosmic rays. Field data consist of the magnetic field, the electric field and the electromagnetic waves generated by local plasma phenomena and by remote sources, especially the Sun. Various instruments are available to perform these measurements. There are missions like Pioneer Venus '78 with limited capabilities using three instruments weighing only 5 kg and also missions like Voyager capable of measuring all the above properties with six instruments weighing almost 40 kg. Thus, when planning future planetary missions, 10 to 25 kg of the science payload should, if possible, be allocated for particle and field instruments.

The locations of the recently discovered gamma ray bursts are determined by triangulation using time of arrival data. Two (or more) detectors on planetary spacecraft can be used to determine accurate source locations for identification with known astronomical objects. Such an instrument need not be large; the Pioneer Venus device is only 2.4 kg.

Conclusions and Recommendations

The most favorable opportunity appears to be the Mercury Orbiter missions with solar/gravity science. It and the other single space-craft opportunities do not require significant advances in spacecraft or propulsion technology. In general, the single launch vehicle oppor-tunities require advanced propulsion systems and/or SEEGA trajectories. Additional study is recommended to determine feasibility of the solar probe mission, the atmospheric probes for the Earth and RPVs for Mars.

MULTIPLE DISCIPLINE SCIENCE ASSESSMENT

1. INTRODUCTION

The planning of most planetary missions is based only upon objectives for planetary science. This report takes a general look at other science disciplines to determine where and when it is appropriate to include them in the planning of planetary missions. Some specific examples of multiple discipline opportunities are then selected and for each a brief description of the mission characteristics is given.

The proper perspective for this effort is set forth below in discussions of previous multiple discipline efforts and of the scope of the current study. The objective of the second section is the identification of promising opportunities for multiple discipline science. A search method based upon science objectives is applied to the cases where a single spacecraft can be used and to opportunities using a single delivery vehicle. Opportunities using a common engineering system are also considered. Each subsection in the third section contains a brief description of the most promising opportunities, including, in most cases, some key mission characteristics and performance data. The final section presents the study conclusions and recommendations.

1.1 Background

Planetary missions have often included an additional science discipline in the form of particle and field instruments. During the interplanetary flight, measurements would be made of the magnetic field, the solar wind, the solar and galactic cosmic rays and micrometeoroids. This has been a useful relationship because data on these fields and particles could be obtained only by _in situ_ measurements and because the planetary mission traverses regions of space where interesting data could be obtained. Planetary science benefits, too, because there is an additional justification for these missions and perhaps additional

financial support as well. The goal of this study is to find additional examples of this type of symbiotic relationship.

Multi-target missions are another way to provide additional justification for planetary missions. The Mariner 10 and the Voyager missions use this technique not only to increase the scientific content but also to gain a performance advantage because of gravity-assisted trajectories. Other multi-target missions are known; for example, Venus-Earth gravity-assisted trajectories and multi-asteroid flybys. While these examples are not multiple discipline opportunities (i.e., all the targets are planetary bodies) they do illustrate the advantages of multiple targets and of gravity-assisted trajectories.

Perhaps the most extensive previous effort to include many science disciplines into a program occurred during the Apollo program. Some Apollo experiments had no lunar objectives and many more had other significant uses in addition to lunar science. The latter included many particle and field type instruments included in ALSEP packages. The former included several experiments with returned materials, including photographs of the Earth, stars, and the zodiacal light. Moreover, the plans prepared for post-Apollo science included even more opportunities for nonlunar science.

1.2 Scope

The stated objective of this study is to identify multiple discipline science opportunities that fit into future planetary missions. At the beginning, a broad understanding of future planetary missions is needed. In the immediate future there are the planetary missions in- cluded in the NASA Five-Year plan for the period FY'80 through FY'84, see Table 2. These missions or some simple variation are described briefly in the Planetary Missions Handbook, Vol. IV [1], where references to the original studies can be found. In addition there is an extensive list of planetary missions concepts which have been studied

Table 2

NASA FIVE-YEAR PLAN FOR PLANETARY MISSIONS

Mission	Project Start	Launch
Venus Orbital Imaging Radar	1980	1983
Comet Rendezvous	1981	1985
Mars Sample Return	1982	1988
Saturn Orbiter/Dual Probe	1983	1987.
Asteroid Multi-Rendezvous	1984	1988

by the Jet Propulsion Laboratory (JPL) and/or by Science Applications, Inc. for which individual references are too numerous to be given here. Not considered here are Earth orbital and lunar missions. The former because they are not for planetary purposes and the latter because it is quite easy to send a mission to the Moon which is dedicated to non-lunar science. (This deletes only one mission, the Lunar Polar Orbiter from the set of possible future missions.)

There are many science disciplines that can profit from observations made in space. The search for those science opportunities which can be accomplished together with the study of planets considered the general disciplines of astronomy, physics, the geosciences and the applied sciences. In astronomy, there is interest in both solar observations and in views of stars and galaxies. The areas of physics which deserve special mention are solar physics, cosmology and gravitational physics. The geosciences and the applied sciences are included because they may benefit from the technological developments needed for future planetary missions.

As with planetary missions, there are so many references for other science opportunities that a detailed list is not given. However, most of the basic concepts for the near-term can be found in one of the Space Science Board reports [2,3], and others for the time period through the year 2000 are contained in the report by the Outlook for Space Study Group [4]. These sources extend well beyond the nominal NASA Five-Year Plans. These general references concentrate on NASA activities within the Office of Space Science (OSS); however, there is no intention to limit this study to OSS opportunities, especially for the opportunities in the geosciences and in the applied sciences.

There are three types of commonality of interest that can unite planetary objectives with the other sciences. The first and simplest is the case of using a planetary mission spacecraft to carry out an experiment for the other discipline. The previously cited example of particle and field observations illustrates this type of commonality well. The second type is the use of a single vehicle to deliver two (or more) spacecraft to their targets. It is not unusual for a single launch vehicle to place several satellites into Earth orbit. However, no attempt has yet been made to send two spacecraft to Earth_escape with a single delivery system. All such opportunities would require a ballistic launch and may also involve common use of low-thrust propulsion during an interplanetary trajectory. The last type of commonality is based upon joint use of a system. The interest here is upon complete systems such as atmospheric probes, rovers, etc., and not on a subsystem such as attitude control, propulsion, or a science instrument.

2. OPPORTUNITIES FOR MULTIPLE DISCIPLINE SCIENCE

2.1 Science Objectives

For each science discipline some specific objectives can be identified as potential multiple discipline opportunities. If this is done in a systematic way, it is likely to uncover most significant opportunities. Ideas for science objectives also come from general and specific plans for future space missions [2-4]. However, the approach which starts with science disciplines can cover all disciplines and is adopted for this study.

A multiple discipline opportunity must have some clear advantage over other methods of accomplishing the same objective. Examples of such advantages are higher spatial/spectral resolution, in situ observations, and reduced time or cost for the development of a new technology or a common system. There must be an advantage because of the relatively large propulsion requirements associated with planetary missions and/or the relatively long time to develop and fly these missions. In general, the competing method is either an experiment in Earth orbit or a system operating on the Earth's surface.

Solar Astronomy

The primary objectives for solar astronomy are to improve spatial and/or spectral resolution of the solar disk and corona and to increase spatial and temporal coverage. (This discipline is closely related to solar physics which is discussed later.) Each area can benefit from deep space measurements. Spatial resolution can be improved by going closer to the Sun. The same is also true of spectral resolution, although the Sun is so bright in most spectral regions that the increased brightness may not be necessary. One particularly significant exception is the flux of neutrons which cannot be observed far from the Sun because they decay naturally into a proton and electron which cannot be distinguished from charged particles from other sources.

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Earth-orbiting instruments for solar observations are available with a wide range of spatial/spectral resolution and spectral coverage. Those with the greatest spatial resolutions are relatively large, typically using an optical telescope with a 1 m diameter mirror. Thus they are not easily taken beyond Earth orbit. High spectral resolution experiments are somewhat smaller, but are usually larger than any instrument on a planetary mission. Consequently only smaller instruments with lower resolutions can be incorporated into a multiple discipline opportunity. Such instruments are not likely to resolve smaller features than an Earth-orbiting experiment, unless they are taken very close to the Sun. In particular, Mercury's orbit (0.31 AU at perihelion) is probably not adequate.

To improve spatial and temporal coverage it is necessary to leave the Earth's vicinity. Specifically, there are uses for views of the Sun's polar regions and views of all regions from different aspects so that vertical structures can be identified with stereoscopic techniques and so that surface features can be followed continuously, not just when that solar longitude faces the Earth. There are three special orbits which can be used to extend spatial and temporal coverage. The first is a circular orbit at 1.0 AU but with a phase angle of about 90° with respect to the Earth. This orbit is ideal for providing the stereo coverage and it also extends temporal coverage somewhat. The second is a circular orbit, again at 1.0 AU but with an inclination of 90°. Its use is primarily for coverage of polar regions. Some extra spatial and temporal coverage is possible depending upon the details of the final orbit. The last special orbit is also circular, near the ecliptic plane at a distance of about 0.20 AU from the Sun so that it has no motion relative to the rotating solar surface. (This orbit is the solar analog of a geosynchronous orbit.) Clearly no planetary orbit is a substitute for one of these special orbits. However, temporal and spatial coverage can be extended by observing the Sun from Mercury. Since Mercury has no atmosphere, it may be used to occult the

solar disk for low background measurements of the solar corona. Venus and Mars do provide different aspects, but achieving high resolution is harder than it is at Mercury.

Stellar and Galactic Astronomy

For stellar and galactic astronomy the primary interest in deep space observations is for lower background or less interference, for in situ measurements, and for increased flux due to the focusing effect of the solar gravitational field. The spatial, spectral and temporal resolution from the Earth's surface or from Earth orbit are not going to be improved upon. Indeed, the large size of the highest resolution instruments cannot be duplicated in a deep space mission. There is one exception, location of transient sources of gamma rays, which is discussed at the end of this section.

The measurement limiting background may be due to zodiacal light (solar radiation reflected from small particles), line emission from ionized gases or local sources of radio noise (i.e., on the Earth or Sun). There is also a problem with the solar wind plasma not being transparent for very low frequency radio waves. To escape these sources of interference, the observations should be made as far as possible from the Earth, the Sun and/or the ecliptic plane. A highly inclined orbit is needed to get away from the zodiacal light and the plasma oscillation The far side of the Moon (when it is dark) is a good place to go for radio astronomical observations without interference from the Earth or the Sun. While these circumstances can be duplicated at other places in the solar system (at Mars, for example), the Moon is the preferred place because missions to it are less difficult.

In situ observations of the interstellar medium require that the spacecraft escape the solar system and in doing so go beyond the region controlled by the solar wind and solar magnetic field. At the same time, the spacecraft also reaches a region where there is less interference due to line emissions from ionized gases. The location of the

boundary between solar and interstellar environment is uncertain. For spacecraft traveling near the ecliptic plane, the boundary is reached most easily by escaping in the direction of motion of the Sun relative to the interstellar medium. While the boundary is likely to be much closer at high solar latitudes, the fact that it is more difficult to send a probe on a highly inclined escape trajectory means that this mission should probably stay near the ecliptic plane.

One way to obtain increased sensitivity for astronomical measurements is to use the Sun's gravitational field to focus energy from a distant object such as the center of the galaxy. A spacecraft can be at the focal point for any particular object, which is at a distance of 20 to 40 AU away from the Sun, if the spacecraft, the Sun and the object are in a straight line. One limitation of this mission concept is that it is difficult to move the spacecraft around in such a way that focused energy from a number of sources can be observed. Neptune is at 30 AU and its orbit period is 165 years. So while it may be convenient to combine measurements at Neptune with those of solar focused energy, the latter only covers an arc of 2.2° per year.

Spacecraft in Earth orbit have detected gamma ray bursts. The detectors do not have much angular resolution, so source locations are determined by triangulation using time of arrival data. The accuracy of the source position determined in this way is greatly increased if the detectors are more widely spaced. Thus, two (or more) gamma ray burst detectors on deep space missions plus those in Earth orbit can be used to determine accurate locations for identifications with known astronomical objects.

Solar Physics

Solar physics depends upon deep space missions for _in situ_ observations of the particles and fields in interplanetary space. These measurements are often performed on planetary missions and since more data would be useful, this cooperation should continue. Efforts are

also needed to extend the region where measurements are made beyond current limits. Specifically, much could be learned inside Mercury's orbit, especially by a probe which goes within several radii of the solar disk. The solar probe would provide important data on conditions in the solar corona and its transition to the interplanetary conditions observed elsewhere. Missions close to the Sun may also be able to give some details on the internal structure of the Sun, particularly the mass distribution. For the same reason, a much improved accuracy for Mercury's orbit should be a goal for an orbiter mission.

There is a planned out-of-the-ecliptic mission and additional missions to study the particles and fields at the higher solar latitudes are needed. An exciting new opportunity for solar physics is the collection of samples of the particles in space by a deployed apparatus on a sample return mission. The particles can be from the solar wind, solar flares, galactic cosmic rays and even small micrometeoroids.

Cosmology and Gravitational Physics

In many ways, space is the ideal laboratory in which to perform experiments in gravity physics. The key experiment is to determine the orbital motion of an object as accurately as possible and then derive the deviations from the Newtonian laws of gravity. This experiment must also determine the quadrupole moment of the solar gravitational field. The object can be Mercury or a drag-free satellite in some orbit near the Sun. Ordinary spacecraft are subjected to orbit perturbations, such as those caused by solar radiation pressure and attitude control thrusters. Mercury's orbit is not now known accurately enough to clearly separate the effects due to the solar gravity field and those due to relativistic gravitational theories.

Other experiments which can be performed using spacecraft with well-known orbits are a test of the principle of equivalence and a determination of the effect of gravity on the propagation of radio waves.

The former asks whether gravitational and inertial masses, including the gravitational binding energy contributions, are identical. This can only be done in a three-body system. By using the Sun-Jupiter system plus a third body (e.g., Mercury), very significant tests can be made. The latter involves measuring the deflection of the radio signal and the change in propagation time as the spacecraft-Earth communications link passes near the solar disk.

Geosciences

Planetary missions do not offer significant opportunities for observations of the Earth. However, it is possible for planetary missions to use exploration systems developed in conjunction with Earth applications. The systems used in previous planetary missions, the flyby and orbiter spacecraft and the soft landers, are very special designs. They are consequently expensive and not well-suited to terrestrial use. On a system level this will probably continue to be the case although standardized spacecraft subsystems are now being incorporated into future planetary spacecraft designs. Things that should be considered are the use of systems like atmospheric probes, penetrators, hard landers, rovers and remotely piloted vehicles (RPV).

These systems operate in the atmosphere or on the surface where significant scientific and/or technical data could be obtained on both the Earth and the planets. For example, there is a need for data on the behavior of the Earth's atmosphere above the altitudes commonly used by aircraft and at all altitudes for all the planets except Mercury and Pluto. In all cases the atmospheric data should include wind speed and direction, temperature, pressure and composition. Of special concern at the Earth are the measurements of some ecologically important quantities such as trace constituents and the radiation budget. All of the above measurements are desired over the vertical paths followed by atmospheric probes and the horizontal ones flown by RPVs. The significant additional uses for RPVs are for magnetic surveys and for high resolution imagery of the surfaces of Mars and the Earth.

With respect to measurements at the surface of the Earth or any other solid body, there are applications for geophysical data and for meteorological information. From a penetrator (or hard lander) a geophysical use common to all solid bodies is monitoring of seismic activity. Penetrators can also be used on planets, satellites or asteroids to characterize the immediate area using imaging, heat flow and soil composition techniques. A system like a rover, which has surface mobility, is useful for many additional investigations. Some possible measurements that would be desired over a local area on any body are imagery, gravimetry, magnetometry, soil properties (composition, electrical conductivity, etc.) and seismic profiles (in conjunction with a fixed seismic source or detector). Another common use of a mobile system is for sample collection with analysis being performed at a central location. In addition there are unique tasks that might be given to a mobile system on Earth such as inspection of pipelines and power lines.

Applied Sciences

There are a number of unique environments that are present in deep space or at planets which offer opportunities for the development and testing of items employing new technologies. These environments include the solar wind and solar flare particles, the high solar fluxes at Mercury, temperature extremes from the hot surfaces of Venus and Mercury, the cold of an outer planet satellite, the 100 bar or more atmospheres of Venus and the giant planets and the strong radiation belts at Jupiter. All of these can be simulated on the Earth. However, when developing materials and systems which must work in these environments, a technology demonstration in the actual environment is often considered essential. Of all the environments mentioned above, the solar wind and solar flare particles are the ones which are encountered by some proposed large systems. Specifically, systems such as solar power stations and their associated manufacturing facilities

would operate in geosynchronous orbit, on the lunar surface or at a libration point where these particles could affect structural and/or electrical properties of key components. It would be possible to study such effects by sending samples along on a sample return mission to a planet. They would be exposed to the appropriate environments during the cruise phase and could be returned to Earth for detailed laboratory analysis. With respect to the other unique environments, there appear to be no similarly significant systems, except for planetary missions, which will be exposed. Furthermore, there is no easy way to arrange for return of the sample to Earth for detailed study.

2.2 Opportunities Using A Common Spacecraft

The simplest way to do multiple discipline science is to combine all objectives into one mission using a single spacecraft. This approach is exemplified by the inclusion of solar physics objectives (i.e., particles and fields) in planetary missions. There are two basic questions which must be asked:

1. Does the spacecraft have excess capability to absorb the additional requirements associated with the additional objectives?

2. Is the trajectory used for the planetary mission satisfactory for other objectives?

Multiple discipline missions are possible if the answers to both questions are yes or, at least, yes if some small, acceptable change is made in the planetary objectives.

To begin the opportunity identification process a list of potential planetary missions is needed. The following list has the next one or two missions to each solar system body including the missions in the NASA Five-Year Plan (see Table 2):

Mercury Orbiter
Venus Orbital Imaging Radar
Venus Lander
Mars Geochemical Orbiter
Mars Surface Sample Return

Asteroid Multi-Rendezvous
Comet Rendezvous
Asteroid/Comet Sample Return
Jupiter Satellite Lander
Saturn Orbiter/Dual Probe
Uranus, Neptune and Pluto Flybys

For most of these a description of planetary science objectives and in-
struments, of spacecraft systems, and of typical trajectories can be
found in Reference [1]. Table 3 lists the distinguishing mission char-
acteristics associated with the science objectives that were identified
in the previous subsection. Can any of the above planetary missions
provide an opportunity for the science objectives from Table 3?

The Mercury Orbiter mission is the best choice for extending spatial
and temporal coverage of the solar disk. The payload for solar astronomy
will be small, but not insignificant--probably less than 50 kg. None of
these planetary missions goes close enough to the Sun for a significant
improvement in spatial resolution of the solar disk; nor do these mis-
sions provide spatial coverage of the Sun at high latitudes. A low
perihelion and/or a high inclination orbit can be achieved using a
Jupiter gravity assist. However, any future planetary mission which
goes by Jupiter will almost certainly be targeted for another planet
such as Neptune or Pluto.

Planetary spacecraft do not have adequate payloads to accommodate
the instruments needed for most astronomical observations. This is
particularly true of optical and radio instruments which could take
advantage of lower backgrounds. A gamma ray burst detector, however,
is a desirable addition which is easily incorporated into any planetary
spacecraft. The Uranus, Neptune and Pluto flyby missions offer the
opportunity for in situ observations of the interstellar medium. The
necessary instrumentation can be added to these spacecraft. The signi-
ficant change in the planetary mission is the desire for a long period
for the interstellar observations which extends the post-encounter
phase of the planetary mission.

Table 3

SCIENCE OBJECTIVES FOR DEEP SPACE MISSIONS

Science Discipline	Science Objective	Distinguishing Mission Characteristics
Solar Astronomy	High spatial resolution of solar disk	Close approach to Sun
	More spatial/temporal coverage of solar disk	High inclination to ecliptic of different longitudes than Earth
	Continuous observation of solar disk	Heliosynchronous orbit (at about 0.2 AU)
Stellar Astronomy	More sensitivity for some astronomical measurements	Distant from interference (e.g., Earth) or at solar focus
	In situ observations of interstellar environment	Great distance from Sun (perhaps not so far at high solar latitude)
	Location of gamma ray bursts	None (all trajectories are useful)
Solar Physics	In situ observations of corona	Close approach to Sun (within several solar radii)
	In situ observations of interplanetary environment	None (all trajectories away from Earth are helpful)
Gravitational Physics	More sensitivity of relativistic gravitational effects	Close approach to Sun
Applied Sciences	Demonstration of advanced technologies	Sample return

All of these planetary missions can contribute to in situ observations of the interplanetary environment because the required additional science payload is small and because useful measurements are obtained on any trajectory. None is able to provide data inside the orbit of Mercury, especially on the transition to the solar corona. The Mercury Orbiter mission can be used to improve the accuracy of Mercury's orbit which has implications for the structure of the Sun and for relativistic gravitational effects.

Surface sample return missions are the only opportunity for the demonstration of advanced technologies. The Mars Sample Return mission is chosen over comet and asteroid missions because it is closer to realization. Samples which have been exposed to the deep space environment, especially the solar and solar flare particles, can be brought back to Earth-based laboratories where measurements can be made. These samples can provide interesting solar physics data as well as applied sciences data about the effects on materials.

Thus, Mercury Orbiter, Mars Surface Sample Return, and Uranus, Neptune or Pluto Flyby are the planetary missions which have unique and significant opportunities for multiple discipline science. In addition, all missions offer opportunities for particles, fields and gamma ray burst investigations. Each of these promising opportunities is discussed in more detail in Section 3.

2.3 Opportunities Using a Single Launch Vehicle

The reason it was difficult to associate a planetary mission with some science objectives in Table 3 was because no spacecraft satisfied the distinguishing mission characteristics. Let us now assume that separate spacecraft are used for the planetary mission and for the missions suggested in Table 3. Can a single launch vehicle be used to deliver both? The answer to this question can be yes, especially when a Tug vehicle is used to go from the Shuttle orbit to Earth-escape

or when a 60 kw Ion Drive low-thrust propulsion system is used for the interplanetary trajectory. It is known that these propulsion systems can do relatively difficult missions [5-7]. Thus, it is probable that two less demanding missions with similar interplanetary trajectories can be done simultaneously.

A close approach to the Sun is a common mission characteristic of several science objectives. The best way to get into a low perihelion orbit is to employ a gravity assist at Jupiter. A Jupiter swingby is also a good way to get a planetary spacecraft to Uranus, Neptune or Pluto, in certain launch years. A planetary mission which orbits Jupiter, such as a satellite lander, cannot be combined with a gravity assist mission because the nominal hyperbolic approach velocities are very different. For these combined missions both ballistic and low-thrust trajectories can be considered.

Several science objectives would benefit from measurements made from a spacecraft which is in a highly inclined, circular, heliocentric orbit. With low-thrust propulsion, it is possible to attain such an orbit using trajectories passing by either Mars or Jupiter. The Mars approach velocity would be low, allowing deployment of an orbiter and/ or a lander. After Mars the low-thrust system is used to gradually increase the orbit inclination. Satisfactory Jupiter approach conditions can be found so that the post-encounter trajectory for the solar probe spacecraft has a 90° inclination and so that the other spacecraft can be a flyby mission to Uranus, Neptune or Pluto. After the Jupiter encounter the low-thrust system is used to circularize the highly inclined orbit at about 1.0 AU.

Continuous observations of one side of the solar disk require a circular orbit at about 0.20 AU. Low-thrust propulsion is needed to reach such an orbit and also to deliver orbiters to Mercury. A common propulsion system may be able to go first to Mercury and then to a 0.20 AU orbit.

Most of the science objectives shown in Table 3 are represented in the potential opportunities identified for either common spacecraft or common launch vehicles. The exceptions are the low background stellar astronomy objective and the solar observations at a different longitude than the Earth. The former probably requires larger instruments than can be accommodated here, and some or all of both can be done without going far from the Earth's orbit.

2.4 Opportunities Using a Common System

The last type of commonality to be investigated is the use of a complete system for both planetary and terrestrial applications. The systems useful for planetary exploration are:

> Spacecraft (orbiters and flyby)
> Landers
> Atmospheric Probes
> Rovers
> Penetrators (or hard landers.)
> Remotely Piloted Vehicles
> Buoyant Stations (balloons).

The first three have already been developed for planetary missions. Thus, new terrestrial uses are sought for these. The others are systems which have been proposed for planetary missions, so the emphasis is on finding a planetary application for a system that is similar to a terrestrial use. If applications can be found using a common system, then the development costs could be shared and perhaps the development time could be reduced. Note that this effort is restricted to complete systems and does not consider separate subsystems such as attitude control, propulsion or science instruments.

Complete spacecraft and landers are not an appropriate opportunity for multiple use, because some requirements for terrestrial and planetary missions generally are very different. For example, planetary spacecraft have much greater communications distances and a wide range of heliocentric distances which impacts the thermal control and power systems. In addition, progress is being made on the subsystem level

where low-cost NASA standard equipment developed for terrestrial use is now found on new planetary spacecraft. The designs of landers are largely determined by the atmosphere, if any, and by the surface gravity of the planetary body. Furthermore, there is no obvious need for a terrestrial system like a lander--any function it could perform is probably easier to do some other, less costly way.

There is a common need on the Earth and on planets for data on atmospheric structure and composition. Terrestrial information has been acquired using balloons and sounding rockets, but atmospheric probes deployed from orbit are an alternate method which may be attractive now that Shuttle has reduced the cost of going into Earth-orbit.

Rovers and penetrators are very useful systems for the exploration of a planetary surface. A rover could also move across the Earth's surface performing geophysical and geochemical measurements or technical tasks such as inspection of pipelines and utility lines. The hardware for Earth applications, however, would probably be very different. In particular, the power source could use hydrocarbon fuels or batteries and the rover could be controlled by a human operator. The penetrators which have been used on the Earth have been dropped from aircraft which is simpler than deployment from an orbiting spacecraft. Planetary applications also require deployment with high reliability over a wider range of surface conditions. The requirements for hard landers are similar, with the added difficulty of little terrestrial experience.

Remotely piloted vehicles (RPVs) and buoyant stations (or balloons) are two ways to maintain approximately constant altitude in an atmosphere where data can be collected on the atmosphere itself and in some cases on the surface. Aircraft and/or satellites can substitute for an RPV or balloon in many terrestrial applications. However, RPVs operating at a pressure of about 5 mbar in the Earth's atmosphere could also operate near the surface of Mars. The airframe, avionics, power plant and instrumentation might be common to both applications.

A buoyant station to operate at 5 mbar is also a possibility, but the technical problems are also considerable and the RPV has more lateral mobility.

On this basis the atmospheric entry probe and the remotely piloted vehicle are selected for further study.

3. CHARACTERIZATIONS OF MULTIPLE DISCIPLINE OPPORTUNITIES

In this section, specific characteristics of missions with multiple discipline science objectives are described. All three types of commonality are considered. First, the impact of additional objectives is studied for three selected planetary missions, namely:

1. Mercury Orbiter
2. Mars Surface Sample Return
3. Neptune or Pluto Flyby.

The impact of including field, particle and gamma ray burst detectors on any planetary mission is also discussed. Second, three opportunities are described which deploy two spacecraft, a planetary mission and one of the following missions:

1. Synchronous Solar Observatory at 0.20 AU
2. Out-of-the-Ecliptic Solar Polar Observatory
3. Solar Probe to 0.02 AU.

Third, two systems, atmosphere probes and remotely piloted vehicles, are proposed which could be developed jointly for planetary missions and for terrestrial applications.

3.1 Mercury Orbiter

Rationale

A spacecraft in orbit about Mercury can acquire unique data on the Sun and on relativistic gravitational effects. The advantages for solar observations are higher spatial resolution, greater solar flux, a longer time for observing individual features and the possibility of using Mercury as an occulting disk for coronal studies. The additional payload available for solar experiments will be modest. Preference is given to instruments which extend spatial and/or temporal coverage, since this cannot be done from Earth orbit. A Mercury Orbiter does not offer a better opportunity for higher spatial and/or spectral resolution. It is probably easier to increase the resolutions of Earth orbital instruments. Observations of solar neutrons should be attempted

since they are much more difficult to observe at the Earth. From a much more accurate determination of Mercury's orbit, information on the internal structure of the Sun can be derived and tests can be made on relativistic gravity theories.

The conditions for solar observations from spacecraft orbiting planets are given in Table 4. At Mercury an optical instrument has between 2.14 and 3.25 times greater spatial resolution of the solar disk than the same instrument in Earth orbit. This means that the instrument for a Mercury Orbiter can have the same spatial resolution as one at Earth even though its angular resolution is about 2.5 times less. Since angular resolution is determined by the diameter of the primary optical element, this dimension can be reduced about 60%, resulting in a smaller, lighter instrument. The higher flux is of little value except for the case of neutrons which, if they decay naturally into a proton and electron, cannot be distinguished from charged particles from other sources. The neutron half-life is about 17 minutes, so many solar neutrons decay before they reach a planetary orbit. For example, a 4.7 MeV neutron moves at a velocity of 10^7 m/sec and requires 96 minutes and 250 minutes to reach Mercury and the Earth, respectively. While only 0.35% survive the trip to Mercury, this is 8,500 times the probability at the Earth. Including the average relative flux factor from Table 4, the solar neutron flux (4.7 MeV or less) at Mercury is more than 50,000 times that at the Earth. The solar occultations by Mercury and Mercury's orbit eccentricity can both be used to demonstrate that observed neutrons are from the Sun.

The times during which observations can be made of features on the solar disk are computed assuming that visible solar features are in a 120° segment centered on the planetary longitude and that features revolve once each 25.4 days. From the Earth a feature is seen for about 9 days during which time the apparent solar rotation is 120°. There are about 18 days before the feature can be seen again. At Mercury the observing time is typically 12 days and as long as 15 days

Table 4

SOLAR OBSERVATIONS FROM PLANETARY ORBITS

Planetary Orbit	Relative Spatial Resolution	Relative Flux	Observing Time[a] (days)
Earth	1.00	1.00	9.1
Venus	1.38	1.91	9.5
Mercury[b]	2.14	4.59	10.5
	2.58	6.67	11.9
	3.25	10.58	15.0

a. Time for an apparent solar rotation of 120°.
b. Conditions at aphelion, mean distance and perihelion.

when Mercury is at its perihelion where it is closer to the Sun and is moving more rapidly. Under the proper circumstances instruments at Mercury can fill the 18 day gap in Earth-based coverage. When the longitudes of Mercury and the Earth are similar, the observing time is not extended much. However, this condition is desired, too, because stereo observations can be made when the Mercury-Sun-Earth angle is not large.

It is well known that the perihelion of Mercury advances at a rate of 43" per century due to either relativistic effects or the solar oblateness. Tracking of an orbiting spacecraft can significantly improve the accuracy of Mercury's orbit. It is possible that these new data can be used to determine the magnitudes of these two separate effects. Two other experiments to be performed are a test of the principle of equivalence and a determination of the effect of gravity on the propagation of radio waves. The former asks whether gravitational and inertial masses, including the gravitational binding energy contributions, are identical. This can only be done in a three-body system. The Sun-Jupiter system plus a third body, Mercury, allows a

most significant test to be made. The latter involves measuring the deflection of the radio signal and the change in propagation time as the spacecraft-Earth link passes near the solar disk.

Instruments

There are many potential instruments which could be used to investigate the Sun from a Mercury Orbiter. Those on the Solar Maximum Mission [9] are typical (see Table 5); the spectral range is near-IR to gamma rays, spectral resolutions are broadband to $\Delta\lambda/\lambda \sim 100,000$ and spatial resolution is as small as 1". Not represented are ground-based techniques which are characterized by high spatial, spectral and temporal resolution at visible wavelengths.

Table 5

INSTRUMENTS FOR SOLAR MAXIMUM MISSION

Experiment	Spectral Range	Spectral Resolution	Spatial Resolution
Gamma Ray Spectrometer	0.3-17 MeV, 10-160 MeV	7.5% (at 0.66 MeV)	Full Sun
Hard X-Ray Spectrometer	20-300 keV	16 Channel	Full Sun
Hard X-Ray Imaging Spectrometer	3.5-30 keV	6 Channel	8" × 8"
Soft X-Ray Polychromator	1.4-22.4 Å	<0.02 Å	10" × 10"
XUV Spectroheliometer	20-716 Å, 920-1336 Å	0.1 Å	4" × 4"
UV Spectrometer and Polarimeter	1100-3000 Å	0.02 Å	1" to 30"
Coronagraph/ Polarimeter	4432-6583 Å	7 Filters	6.4" to 12.8"
Solar Constant Monitoring Package	UV-IR	--	Full Sun

It is suggested that several solar instruments be included on the
Mercury Orbiter. At least one should duplicate a ground-based capa-
bility for study of active regions in the photosphere. An example is
an imaging system with spectrally selective filters such as Hα, cal-
cium K line, etc. Spatial resolution should be the equivalent of 1"
from the Earth. Such an instrument would have support requirements
similar to a high resolution imaging system used for planet observa-
tions. Specifically, the mass would be 15 kg or somewhat more and the
power about 15 w; a tape recorder and a high data rate telemetry chan-
nel would also be needed. Perhaps this instrument could also image
the corona when the Sun is occulted by Mercury. A second instrument
should have high spectral resolution, much like the UV spectrometer
and polarimeter on the Solar Maximum Mission, which can be used to
deduce the temperature, density and velocity of particles injected
into solar flare and corona plasmas as well as the local magnetic
fields. This instrument would probably have support requirements
similar to the imaging experiment. Another potential instrument is
a solar neutron detector. This device might also be designed to de-
tect high energy solar gamma rays. A reasonable instrument could be
constructed which weighs between 10 and 20 kg, consumes less than 5 w
of power and uses a very low data rate. For lower background this
sensor might be located on a boom. Other instruments from Table 5
could be added if a large payload margin remains.

The accurate determination of Mercury's orbit is done by tracking
the spacecraft using the telecommunications system which provides range
and range rate data. For best results the radio system must be at
least equivalent to Voyager and the spacecraft should have no unknown
forces acting upon it while orbit data are taken. This might be accom-
plished with "quiet" periods of spacecraft operation (i.e., no man-
euvers), with accelerometers to measure forces, or with a "drag free"
design that compensates for forces.

The minimum payload is, therefore, about 30 kg and consists either of two small optical systems or one optical instrument and the neutron detector. The maximum solar related payload is taken to be 90 kg, corresponding to three or four optical instruments and the neutron detector. This is also close to the maximum science payload included on typical planetary missions.

Impact on Planetary Mission

Consider first the problem of including these solar instruments on a nadir pointing Mercury Orbiter. The planetology instruments are assumed to be body-fixed. They require pointing toward Mercury and would prefer a circular, polar orbit. It would be possible to simultaneously point other instruments at the Sun if they were on a scan platform (one degree-of-freedom). There would be some conflicts to be resolved in planning orbit operations, particularly those related to data storage, communications, and tracking. Physically, the major alterations would be in the structure to provide the pointing and in the power system to provide the additional power and cabling. It is estimated that these changes will increase the mass of the spacecraft by 15 to 45 kg for additional science payloads of 30 to 90 kg. A significant amount of propellant, 25 to 75 kg, must be added to the retro system used to put the spacecraft into a 500 km, circular orbit after being delivered by a low-thrust propulsion system. The overall weight requirement is, therefore, 70 kg for the minimum additional payload and 210 kg for the maximum.

There is another way to implement this multiple discipline opportunity. The solar instruments can be body-fixed on a spacecraft and if there are instruments which must be pointed at the planet, they would need a scan platform. This spacecraft would be in an eccentric orbit that is preferred by the particle and field instruments which would also be on this spacecraft. Operational conflicts on the spacecraft would be reduced but those on the Earth would remain. The weight

increase in the structure and the power systems would be similar, but because of the eccentric orbit the added propellant would be only 5 to 15 kg. The total additional mass would be 50 kg for the minimum solar science payload and 150 kg for the maximum.

Mercury Orbiter missions are delivered by low-thrust propulsion systems (for details see References [7] and [8]) for which an increase in the required.net mass in orbit can be achieved with a longer flight time. Typically this sensitivity is 0.3 days per kilogram although for the specific cases shown in Table 6, the sensitivities are up to a factor of two different. Thus, the additional flight time for 30 to 90 kg of additional, multiple discipline science is 15 to 45 days in the dual orbiter mode and 21 to 63 days if a single spacecraft is used in a circular orbit.

Table 6

FLIGHT TIME SENSITIVITY TO
INCREASE IN SPACECRAFT MASS REQUIREMENTS

Low-Thrust Propulsion System	Launch Vehicle	Flight Time Sensitivity to a Payload Increase	Reference
SEP (18 kw)	Shuttle/IUS(III)	33 days/100 kg	8
Ion Drive (60 kw)	Shuttle/IUS(Twin)	18 days/100 kg	7

3.2 Mars Surface Sample Return

Rationale

The multiple discipline opportunity that can be easily combined with a Mars Surface Sample Return (MSSR) mission is the return of samples exposed to the deep space environment. An MSSR is a good way to use the sample return concept to study the solar wind, solar flare particles and low energy cosmic rays. Each of these is affected by the Earth's magnetic field, so that to study them or their effects on materials requires that the samples be exposed on a deep space mission. Because of the wide range of techniques available in Earth-based laboratories for detailed analyses of samples, returned samples can provide data that would be difficult, if not impossible, to get using automated experiments in space. Samples on an MSSR mission would also be exposed to solar photon radiation, micrometeoroids, high vacuum and other effects common to all space missions.

The return of samples exposed to the deep space environment is of interest for the data that can be obtained on the environment itself and for the data showing the effects on materials to be used for future space applications. The goal for solar physics investigations would be to provide much greater sensitivity than is otherwise available. This is possible because of the long exposure time on MSSR missions and because of the sophistication offered by Earth-based laboratory analysis. With respect to the applied sciences, the MSSR mission offers an opportunity to expose carefully selected and prepared materials of interest in the design of future systems. While these environments can be simulated in laboratories, when developing materials and systems which must work in these environments, it is often determined that there must be a technology demonstration in the actual environment. Some proposed large systems, such as solar power stations and their associated manufacturing facilities, operate in places (geosynchronous orbit, the lunar surface and the libration points) where the above environments could affect structural and/or electrical properties of key components.

Experiments

The following experiments which were performed during the Apollo missions indicate the extent of previous experience with investigations of this type:

1. Solar Wind Foils
2. Dielectric Track Detectors
3. Surveyor Television Camera.

The solar wind foils had moderately large surfaces ($\sim$0.4 m^2) which were exposed for up to 1 day. The thin foil was unfurled by an astronaut and suspended above the lunar surface where it could trap the solar wind ions which were impinging upon it. The dielectric track detectors were either astronaut helmets, plastics or minerals. A typical experiment had a collecting area of up to 0.01 m^2 and an exposure time between 1 and 10 days. For studies of solar flare particles and low energy cosmic rays, samples were exposed outside the vehicles, but higher energy cosmic ray investigations used samples which were inside the vehicle. The return of a television camera from a Surveyor spacecraft provided samples which were exposed for several years but which were not selected or prepared for scientific studies.

Experience shows that this experiment must provide for the exposure of many different materials, each of which has a significant exposed surface area. The nominal experiment is described by the following parameters:

Number of samples	100
Exposure time	1100^d
Minimum area	10^{-3} m^2
Maximum area	0.1 m^2
Maximum thickness	10 mm
Minimum thickness	0.1 mm
Average weight per sample	50 g

The total number of samples is large; however, a key factor in most of the proposed scientific studies is the variation in the observed

quantities for different surfaces. A typical MSSR mission lasts about three years. This is significant because the results of Earth-based laboratory experiments on these samples can be used for extrapolations with reasonable confidence to the operational lifetimes of large systems and because each sample could collect at least 100 times more particles than an Apollo sample since the areas are similar and the exposure time is at least 100 times greater. The total sample area is estimated to be about 2 m^2 and the total weight is 5 kg. Smaller experiments could be designed using either fewer samples or smaller areas and larger ones with more or larger samples.

It is also recommended that a portion of the MSSR science payload be allocated to a package of field and particle instruments to measure the environments to which the samples are exposed. The essential instruments are:

1. Magnetometer
2. Electric Field Detector
3. Plasma Analyzer
4. Low Energy Particle Detector

which as a group should weigh no more than 10 kg and which make only small demands upon other spacecraft subsystems (see also Section 3.9).

Impact on the Planetary Mission

An estimate of the impact that this additional objective would have was made for a 1988 dual launch MSSR mission. It is assumed that this experiment is launched with the Mars Orbiter spacecraft which includes the Earth return vehicle (ORB/ERV). For this analysis, the 5 kg sample mass is assumed to be part of the Earth Orbit Capsule (EOC) into which the Mars sample is inserted. An additional mass of 6.25 kg is added to both the EOC and the ERV to account for a sample canister and hardware to deploy and retrieve the sample. In addition, the ERV is given an allocation of 20 kg for the field and particle instruments, including all required support. This 37.5 kg increase in

the spacecraft net mass results in a 325 kg increase in the injected
mass requirement for the ORB/ERV. Each additional kilogram of sample
requires about 20 kg more injected mass. For this launch opportunity,
the ORB/ERV margin is about 1700 kg assuming a 1 kg Mars sample, an
.IUS(Twin) and Earth-storable propellants for Mars orbit operations [6].
Additional Mars samples can also be accommodated requiring about 33 kg
`of additional injected mass for the ORB/ERV for each kilogram of sur-
face sample. It is known that the ORB/ERV margins are comparable for .
all launch opportunities between 1988 and 1999 [6], so that this mul-
tiple discipline opportunity does not depend upon selection of a
favorable launch opportunity. It is not expected that a sufficient
margin will exist for this opportunity to be included in a single
launch concept for a MSSR mission.

3.3 Flyby Missions to Neptune or Pluto

A spacecraft on a Neptune or Pluto flyby mission offers an excel-
lent opportunity for study of the interstellar medium and its inter- ·
actions with the solar wind. This boundary is currently estimated to
be 50 AU but this could be in error by as much as a factor of 2. The
planets Uranus, Neptune and Pluto are all presently at ecliptic longi-
tudes where the boundary is relatively close to the Sun. This condi-
tion will persist at least until the year 2000. Thus, this objective
requires observation during the cruise phases of the mission, par-
·ticularly after planetary encounter. The quantities to be observed
are the interstellar magnetic field and the interstellar particles,
both charged and neutral. The appropriate instrumentation is:

1. Magnetometer
2. Plasma Particle Analyzer
3. Cosmic Ray Detector
4. Neutral Mass Spectrometer.

It is presumed that most of the necessary instrumentation is already
included for the purpose of measuring the interplanetary particles and
fields (see Section 3.9).. The obvious additional experiment is a

detector for neutral atoms and molecules. Its impact on the spacecraft is negligible. Some increases may be needed in the sensitivity and/or the energy ranges of some instruments. For example, the interstellar magnetic field is estimated to be 0.1γ. The Voyager magnetometer has sufficient sensitivity, but some improvement is needed in the absolute error which is estimated to be almost 0.1γ. In addition to these particles and field investigations, there is also a potential interest in ultraviolet astronomical observations because the spacecraft will be outside a region of ionized gases which causes a rather strong background at some wavelengths. A UV spectrometer may already be included for measurements of planetary atmospheres.

The inclusion of this objective implies that the mission duration should be as long as possible--limited only by spacecraft reliability or communication capability. Some subsystem modifications may be advised, but they are not expected to add significantly to the weight.

Spacecraft are not easily sent to Neptune and Pluto. Swingby trajectories via Jupiter or Saturn are advisable. Table 7 lists some appropriate launch opportunities and flight times. All opportunities shown have flight times of 8 years or less to the last planet and deliver a 600 kg spacecraft with provision for a 200 kg atmospheric probe for the Neptune cases. (All data are taken from Reference [5].) Providing that a 40 kw solar electric low-thrust propulsion system is developed, Neptune can easily be encountered using launches in 1983 through 1989. The Uranus-Neptune launch opportunity shown for 1987 is typical of opportunities each year from 1985-1989. If this mission is restricted to ballistic trajectories, then a Jupiter swingby must be used if the flight time is to be less than 8 years. The relevant opportunities are 1992 and 1994 for Neptune and 1990 and 1991 for Pluto. Flight time is decreased by using a larger launch vehicle, e.g., a Tug instead of an IUS, but this is not always advantageous because the faster trajectory may have a swingby distance below Jupiter's surface. In 1992 Jupiter-Neptune missions have this problem. In addition,

Table 7

NEPTUNE AND PLUTO MISSION OPPORTUNITIES

Launch Year	Planets Encountered	Launch Vehicle	Time to Last Planet*
1983	Saturn-Uranus-Neptune	Shuttle/IUS(Twin)/SEP (40 kw)	6.9 years
1985	Saturn-Uranus-Neptune	Shuttle/IUS(Twin)/SEP (40 kw)	7.0
1987	Uranus-Neptune	Shuttle/IUS(Twin)/SEP (40 kw)	7.5
1990	Jupiter-Pluto	Shuttle/IUS(Twin/Spinner) Shuttle/Tug(E)/EE-Kick	7.4 6.3
1991	Jupiter-Pluto	Shuttle/IUS(Twin/Spinner) Shuttle/Tug(E)/EE-Kick	7.9 6.4
1992	Jupiter-Neptune	Shuttle/IUS(Twin/Spinner) Shuttle/Tug(E)/EE-Kick	7.9 6.8
1994	Jupiter-Neptune	Shuttle/Tug(E)/EE-Kick	6.7

*Payloads are 800 kg at Neptune (includes probe) or 600 kg at Pluto.

some restrictions on the planetary encounter may be necessary to put the spacecraft on a post-encounter trajectory that escapes the solar system at a rapid rate. Obviously, the fastest trajectory to Neptune or Pluto is desired.

3.4 Mercury Orbiter Missions and Synchronous Solar Observatory

Rationale

A spacecraft in a circular heliocentric orbit at 0.17 AU would have an orbital period of 25.4 days. The rotation period for the solar photosphere is 25.4 days at the solar equator and somewhat slower

at higher latitudes. A spacecraft in such an orbit can continuously observe the same solar longitudes and, therefore, see the birth, development and decay of features on the solar disk, in the solar corona and in the solar wind. For convenience, this orbit is called the synchronous solar orbit and the 0.20 AU orbit. Since this orbit is inside Mercury's orbit, a Mercury Orbiter mission can be delivered first. Then, the low-thrust propulsion system can take the solar observatory to 0.20 AU.

The solar observing conditions for this orbit are (see Table 4 for comparative data from planetary orbits):

Relative Spatial Resolution	5.00
Relative Flux	25.00
Observing Time	∞

This means that excellent spatial resolution, equivalent to 1" from the Earth, is easily obtained with modest instrument optics. Instruments with high spectral resolution and wide spectral coverage should be considered so that full advantage can be taken of the essentially infinite time available for observing individual features on the Sun. The Earth-Sun spacecraft angle changes constantly, so there will also be opportunities for stereoscopic observations of the photosphere and the corona.

For a 4.7 MeV neutron, the travel time to 0.20 AU is 50 minutes (see the discussion in Section 3.1), hence 5.3% of these neutrons reach 0.20 AU. A detector at 0.20 AU sees a flux of 4.7 MeV neutrons that is about 60 times greater than the flux at Mercury's mean distance and 3 million times greater than at 1.0 AU. Since the spacecraft stays at a constant longitude, this mission offers an excellent opportunity to study the time dependent behavior of the solar particles and fields. The position at 0.20 AU is closer to the Sun than spacecraft have been heretofore, so this mission does extend the range of heliocentric space which has been studied. However, the environment at 0.20 AU is expected to be easily predicted based upon current data down to 0.30 AU. (The Solar Probe mission discussed in Section 3.6 extends the explored

region in to 0.02 AU.) This mission can also be used for studies of relativistic gravitational effects; the scientific objectives are the same as those described for Mercury orbiters (Section 3.1). It will be possible to take data simultaneously from two spacecraft, one orbiting Mercury and the other at 0.20 AU. When reduced together, it may be easier to separate relativistic effects from orbit perturbations due to the solar mass distribution.

Instruments

The candidate optical instruments are similar to those used on the Solar Maximum mission (shown in Table 5) plus a capability similar to an Earth-based observatory. From such a list, three or four instruments, weighing about 75 kg, would be selected for the minimum payload. The minimum payload would also include a solar neutron detector (10 kg) and an appropriate set of field and particle instruments (about 15 kg-- see Section 3.9). The total minimum payload is about 100 kg, requires about 100 w of power, and needs a tape recorder and a high data rate telemetry channel.

A possible list of optical instruments for the minimum payload is:

1. An imaging system for photospheric studies at visible wavelengths with spatial resolutions same as 1" from Earth and with selective filters such as Hα and calcium K line.

2. A visible spectrometer/polarimeter with very high spectral resolution (0.02 Å) for studies of the temperature, density and velocity of particles injected into flares and the corona as well as the local magnetic field.

3. A UV spectrometer/polarimeter with similar resolution for similar purposes.

4. A coronagraph/polarimeter for studies of the extended corona.

The first two instruments would duplicate Earth-based capabilities and the last two would have capabilities similar to the Solar Maximum mission.

It is expected that the solar neutron detector could also detect gamma rays although a separate instrument may be needed to obtain the best spectral resolution. Accurate determination of the spacecraft orbit is also needed to produce important scientific results. The spacecraft systems needed are discussed below.

The Synchronous Solar Observatory Spacecraft

In many ways the spacecraft used for the solar observatory at 0.20 AU would be similar to the spacecraft used for either the Solar Maximum Mission or for planetary missions. The spacecraft should be able to point its optical instruments at the Sun not only while it is in the desired orbit but also when the low-thrust propulsion system is delivering the spacecraft to that orbit. The orientation of the spacecraft is expected to be controlled by the direction of the acceleration vector for the low-thrust propulsion system. If this is the case, it means that the optical instruments must be able to rotate about an axis perpendicular to the Sun and to the acceleration vector. This can be done by placing the instruments on a one-degree-of-freedom scan platform, or by allowing the entire spacecraft including the body-fixed instruments to rotate relative to the low-thrust propulsion system. An alternative is to body-fix the instruments perpendicular to the thrust vector, since after leaving Mercury only tangential thrust is needed to get near the desired orbit.

The high spatial resolution instruments need an accurate and stable attitude control system. It is expected that low-cost NASA standard equipment can meet these requirements in conjunction with fine pointing by the scan platform. The field and particle instruments should have a capability for changing their orientation with respect to the Sun. The high gain antenna used for communications must be pointed at the Earth, so it must have a two-degree-of-freedom mounting. After arriving at the desired orbit, the low-thrust propulsion system can be retained so that it can continue to provide power

for the spacecraft and to assist with attitude control. The spacecraft and the low-thrust propulsion system also must be carefully designed to handle the severe thermal environment at 0.20 AU. Excess thermal energy can be radiated into space by large, high emissivity surfaces which are constantly shielded from sunlight. For surfaces which must receive solar illumination, the absorbed thermal energy is minimized by using small areas and special surface coatings. A thermal shield (much like an umbrella) can be used to shade a sensitive part of the spacecraft. The temperature of the shade and of the solar cells can be reduced by using low angles of incidence. The detailed design must determine that these technical means are sufficient to keep the spacecraft operating temperature within a reasonable range even though the spacecraft thermal environment changes by a factor of 25 between launch at 1.0 AU and the desired orbit at 0.20 AU.

A second significant technical problem exists if the orbit of the spacecraft is to be used for scientific purposes. All spacecraft are subject to nongravitational forces which must be either known or compensated for if the orbit is to be known with sufficient accuracy that useful science can be done (see Section 2.1). When the low-thrust propulsion system is retained in the desired orbit, the nongravitational orbit perturbations caused by solar radiation pressure are relatively large. Thus, making the spacecraft into a "drag free" object should be the better approach. To accomplish this, the spacecraft needs a special subsystem which contains a small test mass that is free to follow an inertial orbit. The subsystem senses the spacecraft motion relative to this test mass and maneuvers the spacecraft so that both follow the same orbit. Hopefully, the low-thrust propulsion system which has been retained can be used for this purpose.

Neither of the above technical problems is expected to cause a major increase in the spacecraft mass compared to other deep space missions. Thus, a spacecraft with an estimated net mass of 600 kg (excluding the power and propulsion functions provided by the Ion Drive)

should be able to support the minimum science payload of 100 kg. A larger science payload (200 kg) would require a spacecraft weighing about 800 kg. These estimated spacecraft weights are now used to analyze the performance of a low-thrust propulsion system for the combined missions.

Low-Thrust Propulsion Performance for Dual Mission

For these combined missions, four Mercury Orbiter options were considered. The orbits and required net masses at rendezvous with Mercury are:

Option	Orbit	Net Mass
1. Single Orbiter	Elliptical	600 kg
2. Single Orbiter	Circular	1200 kg
3. Dual Orbiters	1 Elliptical, 1 Circular	1800 kg
4. Orbiter with Three Rough Landers	Circular	2600 kg

Two options were considered for the low-thrust propulsion system. The first is the 60 kw Ion Drive system proposed for Comet Halley Rendezvous. For inbound missions, the solar concentrator is not needed, so the propulsion system mass is 1350 kg. For the performance calculation, the initial power is taken to be 52.8 kw (includes a 12% degradation), the specific impulse is 5000 sec and the overall efficiency is 72%. The other system is a 30 kw solar electric propulsion system using the technology for the proposed Halley/Tempel-2 Rendezvous. Propulsion system mass without concentrators is 1365 kg, initial power is 24.7 kw, specific impulse is 2850 sec, and efficiency is 59%.

The payload to 0.20 AU is shown in Figure 1 for the 60 kw Ion Drive system as a function of flight time and of mass deployed at Mercury. Flight time to Mercury is shown as dashed lines. Results are shown for two Shuttle-based launch vehicles which can be used for Earth
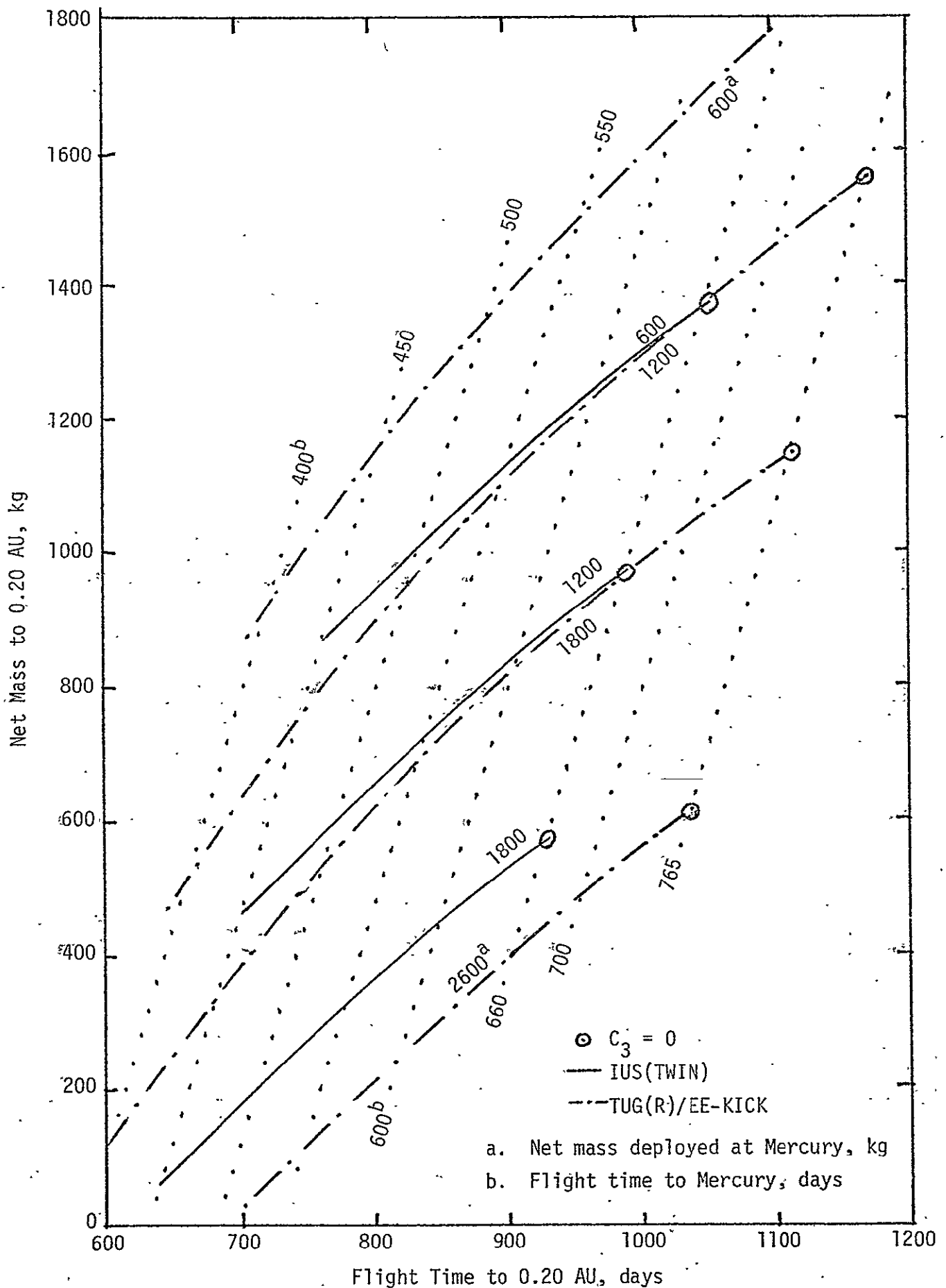
Fig. 1.   NET PAYLOAD TO MERCURY AND TO 0.20 AU USING ION DRIVE

escape. Note that this figure is based on low-thrust trajectories calculated using continuous tangential thrust. Mercury's orbit inclination and orbit phasing are not included; however, results for Mercury alone are close to previous, more exact calculations [7,8]. This approach is used because, unlike other methods, it can be used for the multirevolution trajectories needed to reach a semimajor axis of 0.20 AU.

While the flight times are longer than for either mission alone, it is possible to have many combinations. For example, the single circular orbiter for Mercury (1200 kg) and a 600 kg solar observatory require flight times of 500 and 775 days, respectively, when an IUS(Twin) is used, but only 420 and 690 days when a Tug(R)/EE-Kick is used. Alternatively, the Tug(R)/EE-Kick allows the same Mercury mission (1200 kg with a 500 day flight) to be performed in conjunction with a 950 kg solar observatory requiring an 830 day flight time. Dual orbiters or rough landers for Mercury can be included when the Tug(R)/EE-Kick is used for longer flight time missions.

The 30 kw SEP system is unable to perform this dual mission. Briefly, the results are as follows:

| | | Rendezvous Mass, kg | | Flight Times, days | |
Launch Vehicle	C3	Mercury	0.20 AU	Mercury	0.20 AU
IUS(Twin)	0	600	110	1000	1445
Tug(R)/EE-Kick	0	600	445	850	1235

Not only are the payloads smaller than desired, but the flight times and thruster operating times are too long. It is possible, however, that reasonable flight times and payloads can be achieved by a system with less than 60 kw of power, but with much of the technological sophistication of the Ion Drive system.

3.5 Mission to Planets and A Solar Polar Observatory

Rationale

The out-of-the-ecliptic missions considered here result in highly inclined (>50°) circular orbits. Low-thrust propulsion is required to achieve such an orbit. The objectives of such a mission are to study the structure of the Sun and the properties of interplanetary space as functions of solar latitude. Secondary objectives are stereoscopic solar imagery and low background astronomical observations. The minimum inclination of 50° ensures that remote sensing measurements can be performed easily for all solar latitudes including the polar regions. In situ field and particle data can be obtained over a wide range of latitudes, although a 90° inclination would be preferred for this objective. The circular orbit means that the high latitude portions of the mission will occur regularly. A long duration mission should be considered, perhaps including an active and an inactive part of the 11 year solar cycle. For a circular orbit, the spacecraft is always at the same distance; this simplifies instrument design and data interpretation.

Such a mission would be a logical follow-on to the proposed Out-of-Ecliptic (OOE) mission [10], for which a February 1983 launch is planned. The OOE mission uses a ballistic trajectory with Jupiter swingby to achieve a final orbit with an inclination near 90°, a perihelion of 1.0 AU and an aphelion beyond 5.0 AU. The OOE mission has two spacecraft, each passing over both polar regions during a 1500 day (4.1 year) mission. Subsequent polar passes occur after a full orbit period of at least 5 years. In many ways, therefore, the difference between the OOE mission and the mission described here is similar to the distinction between planetary flybys and orbiters.

The low-thrust propulsion system could be used to slowly increase orbit inclination while keeping a circular orbit at 1.0 AU. However, multiple discipline opportunities exist, with Mars or Jupiter being intermediate targets. An orbiter can be delivered at Mars or an outer planet flyby spacecraft can be separated prior to Jupiter encounter.

Instruments

Because this mission would be a second generation out-of-the-ecliptic flight, the plans should provide for a relative large and sophisticated science payload. It is expected that the remote sensing instruments for solar astronomy would have spatial and spectral resolutions comparable to those for the Solar Maximum Mission (SMM) (see Table 5). For this mission, these resolutions must be achieved at distances of 1.0 to 1.5 AU from the Sun. This means that the instruments for this mission must have primary optics similar to those for SMM.

A possible list of remote sensing instruments is:

1. An imaging system for photospheric studies at visible wavelengths with angular resolution of 1" and with selective filters such as Hα and calcium K line

2. A visible spectrometer/polarimeter with very high spectral resolution (about 0.02 Å) for studies of temperature, density and velocity of particles injected into flares and the corona and for measuring the local magnetic field

3. A UV spectrometer/polarimeter with similar capabilities

4. A coronagraph/polarimeter for studies of the extended solar corona.

The first two instruments are duplications of Earth-based capabilities while the latter two represent Earth-orbital techniques. Some smaller instruments with less spectral/spatial resolution, but with wider spectral range, should also be considered. Note that no specific instruments are identified for low background astronomical observations, although some solar instruments may be used in this mode. The estimated weight of these instruments is 170 kg. They would require about 150 w of power, a tape recorder and a high data rate telemetry channel.

In addition to the remote sensing instruments, the science payload should contain about 30 kg of field and particle instruments (see Section 3.9).

The Solar Polar Observatory Spacecraft

The spacecraft must work together with the low-thrust propulsion system and with the science payload. The low-thrust propulsion system can provide power for the spacecraft and its thrust can assist with attitude control. In turn, the spacecraft may be expected to provide the propulsion system with commands to control the thrust vector, regulated power and a path for sending engineering telemetry back to the Earth.

To support the science payload, the spacecraft must provide pointing, power and commands to the instruments, collect the scientific data and communicate it back to Earth. Of these tasks, the most difficult is instrument pointing. The required accuracy and stability is achievec by the Solar Maximum Spacecraft [9], but the addition of the low-thrust propulsion system results in a spacecraft with much larger moments of inertia and with less structural rigidity. It is also desirable, but perhaps not necessary, that the spacecraft be able to point instruments at the Sun for any orientation of the thrust vector. The instruments can be pointed toward the Sun by a scan platform which has one-degree-of-freedom about an axis perpendicular to both the thrust direction and the Sun. The scan platform must also permit fine pointing to compensate for an expected error in the spacecraft orientation of 0.5°. A tape recorder and a high data rate telemetry channel via a steerable high gain antenna are needed to handle the expected science data.

It is expected that these functions can be easily performed by a SMM or planetary mission subsystem with little impact on subsystem design. In fact, most can be done with low-cost NASA standard equipment. The total spacecraft mass is estimated to be 800 kg, including 200 kg for science, but excluding the functions provided by Ion Drive.

Low-Thrust Performance Including a Mars Orbiter

.Possible functions for a Mars Orbiter mission are remote sensing measurements with emphasis on geochemical properties, deployment of penetrators and support for landed payloads such as rover and sample return missions. In all cases, the basic orbiter (excluding propulsion) is estimated to have a mass of about 575 kg, including up to 100 kg for science instruments. The mass of three penetrators, including deployment mechanisms, is estimated to be 225 kg. It is assumed that the approach velocity is low, so that a propulsion capability of 1500 m/sec is adequate both for insertion into a 1000 km circular orbit (1100 m/sec) and for maneuvers in Mars orbit. The resulting spacecraft mass including propulsion is 1200 kg for the basic orbiter and 1600 kg including three penetrators.

Mars opportunities which could be used for this dual mission occur in the spring of 1986 or 1988 and the summer of 1990. (Low-thrust propulsion will not be available for a 1984 mission.) For the Mars transfer, the low-thrust performance is estimated on the basis of a launch to C3 = 0 and an optimum transfer to Mars with a low arrival velocity (VHP $\sim$ 0) which requires propellant equal to about 10 % of the injected mass [7]. After leaving 1200 or 1600 kg at Mars, the low-thrust propulsion system is used to increase the orbit inclination. Figure 2 shows the net spacecraft mass that can be taken to any particular inclination. These results are based upon a simple formula [11] for the inclination in radians, as a function of the propellant mass fraction, B, namely:

$$i = q \frac{c}{v} [-\ell n (1 - B)] + 0.032$$

where q is a constant (0.827 for a 0.67 duty cycle), c is the exhaust velocity (49 km/sec), v is the heliocentric velocity of the spacecraft (24 km/sec) and 0.032 is the inclination of Mars' orbit. There is also an approximate relationship between B and flight time in days, namely:
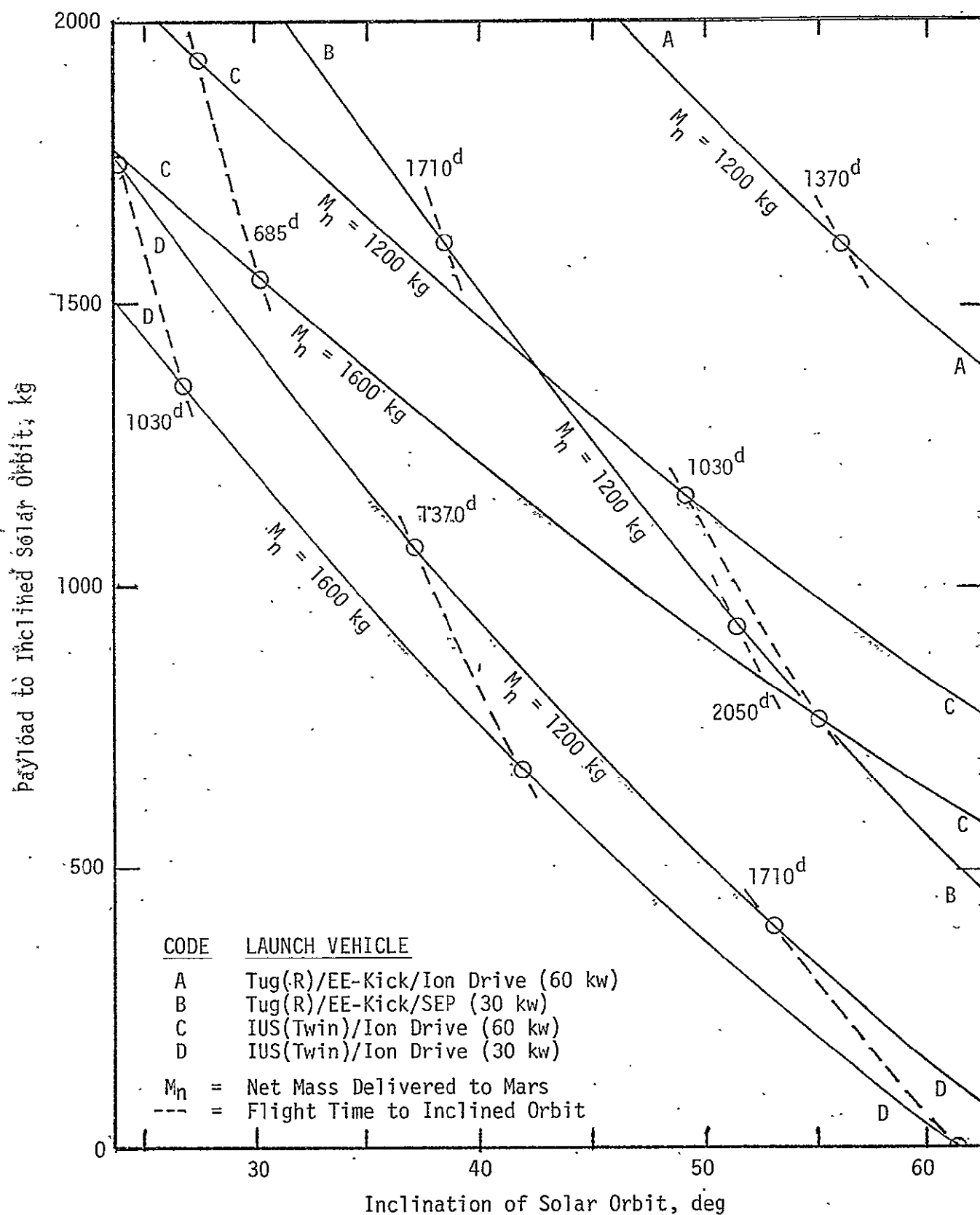
Fig. 2. PAYLOAD TO INCLINED SOLAR ORBIT AT 1.52 AU AND TO MARS

$$B \simeq 4.4 \times 10^{-3} T_F$$

which has been used to plot the flight times on Figure 2. The inclination is not always increasing because thrust is not used during the third of the orbit period when the spacecraft is closest to the ecliptic plane and where thrust cannot easily change orbit inclination. The net mass does not include the mass of the propulsion system; the flight times shown include 225 days for the Earth to Mars transfer. Note that the Solar Polar Observatory is in a circular orbit at 1.52 AU with a 1.88 year period. The larger orbit does reduce spatial resolution of the Sun, but does not seriously degrade the value of the science data.

First we consider the performance of the 60 kw Ion Drive system, which with its concentrator has a mass of 1700 kg. If the IUS(Twin) is used for injection, a payload of 1600 kg can be delivered to Mars and an 800 kg payload can achieve an inclination of 53° in about 1000 days. Using the Tug(R)/EE-Kick launch vehicle increases the mass delivered to Mars ·by about 1100 kg. It is not recommended that this be added propellant for the Solar Polar mission because, although the ultimate inclination is higher, the buildup of inclination is slower. The total flight time to 50° becomes 1250 days. Perhaps a better use of the 1100 kg would be for direct Mars entry of a landed system such as a small rover.

For the 30 kw SEP system, the payloads are smaller and the flight times to inclined orbits are longer. For example, the Tug(R)/EE-Kick case can deliver 1200 kg to Mars and 1000 kg to a heliocentric orbit with a 50° inclination, but the flight time to the latter is over 2000 days. It is not possible to increase the Mars payload or to use a smaller launch vehicle and still deliver at least 800 kg to a 50° orbit.

Low-Thrust Performance Using A Jupiter Swingby

A Jupiter swingby makes it possible for the Solar Polar Observatory to achieve a true polar orbit with 90° inclination. After Jupiter

encounter, the low-thrust propulsion system would be used to circularize the orbit of the Solar Polar Observatory at 1.0 AU. This approach also offers the possibility of separating a spacecraft prior to Jupiter encounter which would be targeted for a swingby trajectory to Uranus, Neptune or Pluto. The favorable launch opportunities to these planets in the 1990's are [5]:

Uranus	Dec 1992, Jan 1994 and Feb 1995
Neptune	Dec 1992 and Jan 1994
Pluto	Oct 1990 and Nov 1991.

First we consider the case of the 60 kw Ion Drive low-thrust propulsion system. Flight time for the Earth-Jupiter trajectory is about 550 days, and the VHP at Jupiter is 13.8 km/sec, resulting in flight times to the outer planets as shown in Table 8. The net mass delivered to Uranus or Neptune is assumed to be about 800 kg which is adequate for a flyby spacecraft and a small atmospheric probe. For a Pluto mission, the spacecraft without an atmospheric probe would be 600 kg.

Table 8

OUTER PLANET FLIGHT TIMES FOR MULTIPLE DISCIPLINE
OPPORTUNITIES USING JUPITER SWINGBYS

Planet	Opportunity	Flight Time*
Uranus	1992	5.3 years
	1994	5.7
	1995	8.2
Neptune	1992	8.0
	1994	9.4
Pluto	1990	7.8
	1991	8.9

*For VHP = 13.8 km/sec at Jupiter

In Figure 3, the net mass to a 1.0 AU circular orbit with a 90° inclination is shown as a function of flight time. It is easily seen that the Tug(R)/EE-Kick is the better launch vehicle for the combined missions. The net mass for the Solar Polar Observatory is 640 kg for the Uranus or Neptune mission and 760 kg for the Pluto missions. Flight time to the circular polar orbit is about 1300 days and that to the planets is shown in Table 8. These payloads could be increased to the desired 800 kg by increasing the flight time beyond flight times considered in Figure 3 or by going to a circular (or elliptical) orbit with a semimajor axis larger than 1.0 AU. (Note that some additional payload may be gained by optimizing the Earth-Jupiter trajectory and by optimizing various Ion Drive system parameters.)

Because the 60 kw system is barely able to perform this dual spacecraft mission, it is unlikely that a smaller system (e.g., 30 kw) can be used. Possibly a SEEGA (Solar Electric Earth Gravity Assist) trajectory, which adds 800 days to the flight time, could be used to give the necessary arrival velocity at Jupiter.

3.6 Missions to Planets and A Solar Probe

Rationale

Another interesting mission for investigations of the Sun is the Solar Probe mission which goes to a perihelion of 0.02 AU. This can be done using a Jupiter swingby where again there is the opportunity to send flyby spacecraft to Uranus, Neptune or Pluto. These combined missions can be done with ballistic trajectories. In this case, the Solar Probe mission probably would consist of a single close passage by the Sun, since the period of orbit is more than 5 years. Low-thrust propulsion, therefore, can be considered as a way to reduce the orbit period and increase the number of perihelion passes.

The choice of perihelion allows _in situ_ study of the solar corona at 4 solar radii, yet offers reasonable hope for a technical solution to the thermal control problem. Significant measurements in the corona
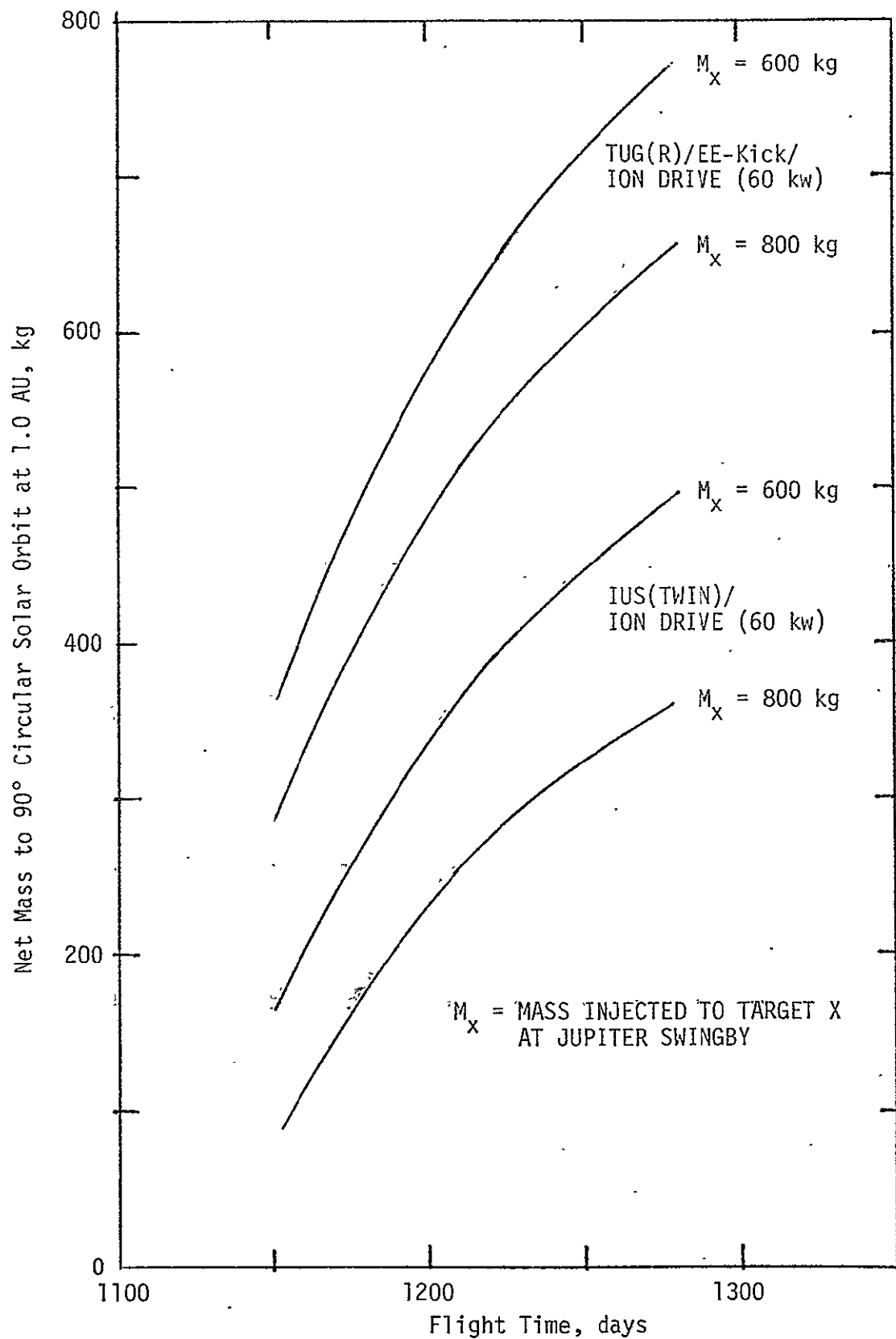
Fig. 3.   PAYLOAD VERSUS FLIGHT TIME FOR SOLAR POLAR OBSERVATORY

are the local magnetic field, composition, charge state, and velocity distribution of the plasma and the energy spectra of the energetic particles. The spacecraft can also observe some of these conditions using remote sensing methods, particularly measurement of radio noise emission, Faraday rotation of the telemetry signal and line emission for ionized atoms. Another objective is to study the solar wind, including the transition from the corona and the significant changes that are expected in the solar wind as it leaves the source region.

This mission also provides an excellent opportunity to measure the solar magnetic field, to detect solar neutrons, to sample small interplanetary dust particles that spiral toward the Sun and to measure the perturbations in the spacecraft motion and radio signal that are caused by relativistic gravitational effects or the solar gravitational potential.

Instruments

The experiments which should be considered for a Solar Probe mission are:

1. Magnetometer
2. Plasma Particle Analyzer
3. Energetic Particle Detector
4. Solar Neutron Detector
5. Plasma Wave Detector
6. Radio Emission Detector
7. Dust Particle Impact Detector
8. UV Spectrometer and Polarimeter
9. V, UV Photometer/Polarimeter
10. Radio Tracking.

The instruments used to measure fields and particles in either interplanetary space or in planetary magnetospheres are generally adequate for the Solar Probe mission. The optical instruments listed are expected to be based upon proven concepts for remote sensing measurements of the solar corona and of light reflected from interplanetary dust. The more sophisticated remote sensing techniques involving imagery and extended wavelength coverage are considered to be less important scientifically and also beyond the spacecraft payload capability. In

designing instruments, careful consideration would have to be given to dynamic range and to potential thermal problems.

The strength of the magnetic field at the solar disk is estimated to be about 1 Gauss. While this is more than 10^4 times the interplanetary field near the Earth, it is similar to the Earth's surface field; thus instruments are available with the proper dynamic range. The magnetometer must be located on a boom away from the spacecraft, but in the antisolar direction so that the instrument is not exposed to direct sunlight. Charged particle detectors with appropriate dynamic ranges are also available. Thermal problems are not easily solved, however. Data are desired on particles moving directly away from the Sun. At a perihelion of 4 solar radii, the disk is in the field-of-view for angles less than 15° from the center of the Sun. Measurements should be made at larger angles outside the solar disk. Some energetic particle data can be taken through a thin thermal shield. For plasma measurements perhaps an electric or magnetic field could be used to divert particles into a detector which is located behind the thermal shield. The solar neutron detector can function adequately with a small amount of material for a thermal shield between the detector and the source of particles.

Flight-qualified plasma wave and radio emission instruments, such as those used on Voyager, are adequate. A dipole antenna is needed for these experiments. It will not be possible to put an efficient (i.e., long) antenna behind the thermal shield. The design of the antenna should attempt to maximize the temperature at which it can survive. When that temperature is reached, the antenna should be folded back into the shaded area. Perhaps a secondary antenna, which is less efficient but protected from direct sunlight, could be used to continue measurements close to the Sun.

The two potential optical instruments are for observations of the corona and are not intended for observations of the solar disk. To do otherwise would cause a significant thermal control problem. It is

expected that simplified versions of existing instruments can be used, such as those for the Solar Maximum Mission described in Table 5.

A radio tracking experiment usually requires no equipment other than the spacecraft radio receiver and transmitter; however, the objectives of this mission demand accurate knowledge of the spacecraft orbit. There are two technical problems which must be solved. First, to reduce nongravitational accelerations, consideration should be given to a "drag free" design for the spacecraft (see Section 3.3). The second problem is choosing radio frequencies and an encounter orientation that result in sufficient signal-to-noise and in the ability to determine the effects that plasmas have on the signal.

It is expected that the minimum science payload of 50 kg would consist of most field and particle instruments plus at least one optical device. The maximum payload is estimated to be 100 kg. All ten experiments cited above would be represented and in some cases a more sophisticated instrument could be included.

The Solar Probe Spacecraft

The spacecraft for the solar probe mission has one major technical problem--thermal control. For each design decision which must be made, the selected option will probably always be the one that is better suited to the severe thermal environment at about 0.02 AU. The spacecraft described here is based upon a JPL concept for such a mission [12].

A concept for the thermal control is shown in Figure 4. The conical heat shield is designed to have a heat radiating area at least twice as large as the collecting area. The candidate materials to survive the expected temperature of about 2000°K are refractory metals, ceramics and graphite. Behind the heat shield are two or more high temperature metallic radiation shields which behave much like the silvered walls of a vacuum bottle and reduce the heat transfered to the spacecraft bus. A blanket of multilayer insulation is attached to the

MAGNETOMETER

RTG

CONICAL
HEAT
SHIELD

SUN

ANTENNA, FOLDED

SPACECRAFT BUS

MULTILAYER INSULATION
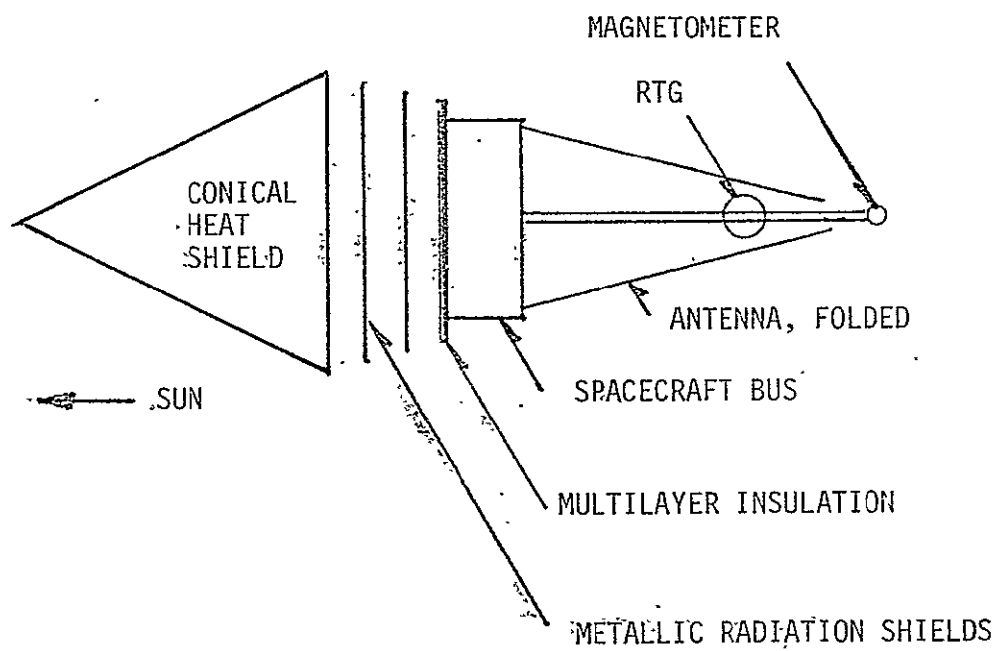
METALLIC RADIATION SHIELDS

Fig. 4. CONCEPT FOR A SOLAR PROBE

spacecraft. More than half of the spacecraft area can radiate excess heat directly into space. The key feature of this concept is that the axis of the conical heat shield must point at the Sun. It is assumed that all science instruments and spacecraft components are in the shade provided by the heat shield. There is a complementary problem, keeping the spacecraft warm at up to 5 AU from the Sun, which must be solved using techniques which have worked on other mission, or perhaps a heat pipe from the RTG.

The spacecraft can either be spin-stabilized or three-axis-stabilized, but any rotation must be around an axis pointing toward the center of the Sun. In either case, it is necessary to provide a capability for a science instrument pointing about an axis perpendicular to the allowed spin axis. On a spinning spacecraft, it would be necessary to despin the high gain antenna and to provide for a rapid change in the spin axis near orbit perihelion. The spacecraft must be powered by RTGs and it must have subsystems for communications, control of operations, etc. A small propulsion system is needed for attitude control and to compensate for nongravitational accelerations.

For science payloads of 50 and 100 kg, the resulting spacecraft masses are estimated to be 600 and 800 kg, respectively. About 25% of this is the mass of the components required for thermal control, principally the heat shield.

Performance Using a Jupiter Swingby

Both ballistic and low-thrust trajectories can be considered for these opportunities. The nominal ballistic missions are easily done by the Tug(E)/EE-Kick with flight times of about 2.4 years to the Sun and 7.2 years to either Neptune or Pluto. Table 9 has flight times for the best Jupiter swingby opportunities to Neptune (in 1992) and Pluto (in 1992). A one year delay results in more than a year longer trip to Neptune or Pluto (see Table 8). The solar flight time is nearly independent of launch year, orbit inclination and orbit perihelion.

Table 9

BALLISTIC MISSION PERFORMANCE FOR SOLAR PROBE
PLUS OUTER PLANET FLYBY MISSIONS

Required Injected Mass[a] (kg)	Flight Times (years) to		
	Sun[b]	Neptune[c]	Pluto[d]
1200	2.2	–	6.8
1400	2.3	6.9	7.1
1600	2.4	7.2	–

a. Solar Probe mass is 600 kg minimum, 800 kg maximum. Flyby
    is 600 kg plus 200 kg for atmospheric probe at Neptune.
    Total required injected mass is sum of these components.

b. Average Jupiter Swingby [12].

c. 1992 Jupiter Swingby [5].

d. 1990 Jupiter Swingby [5].

This Tug vehicle can do both missions and deliver more payload in less
time than can be done for any single target by a single IUS vehicle.

After Jupiter encounter, the low-thrust propulsion system can be
used to reduce the orbit period. The performance of the 60 kw Ion Drive
vehicle is shown in Figure 5. The final orbit period is between 1.2 and
2.0 years, depending on the spacecraft mass and the launch vehicle. A
flight time to the Sun of about 3.0 years gives the maximum payload.
The spacecraft is separated from the low-thrust propulsion system about
25 days before perihelion at a distance of 0.83 AU from the Sun.. This
nominal separation point does not call for operation of the Ion Drive
too close to the Sun and allows sufficient spacecraft tracking prior to
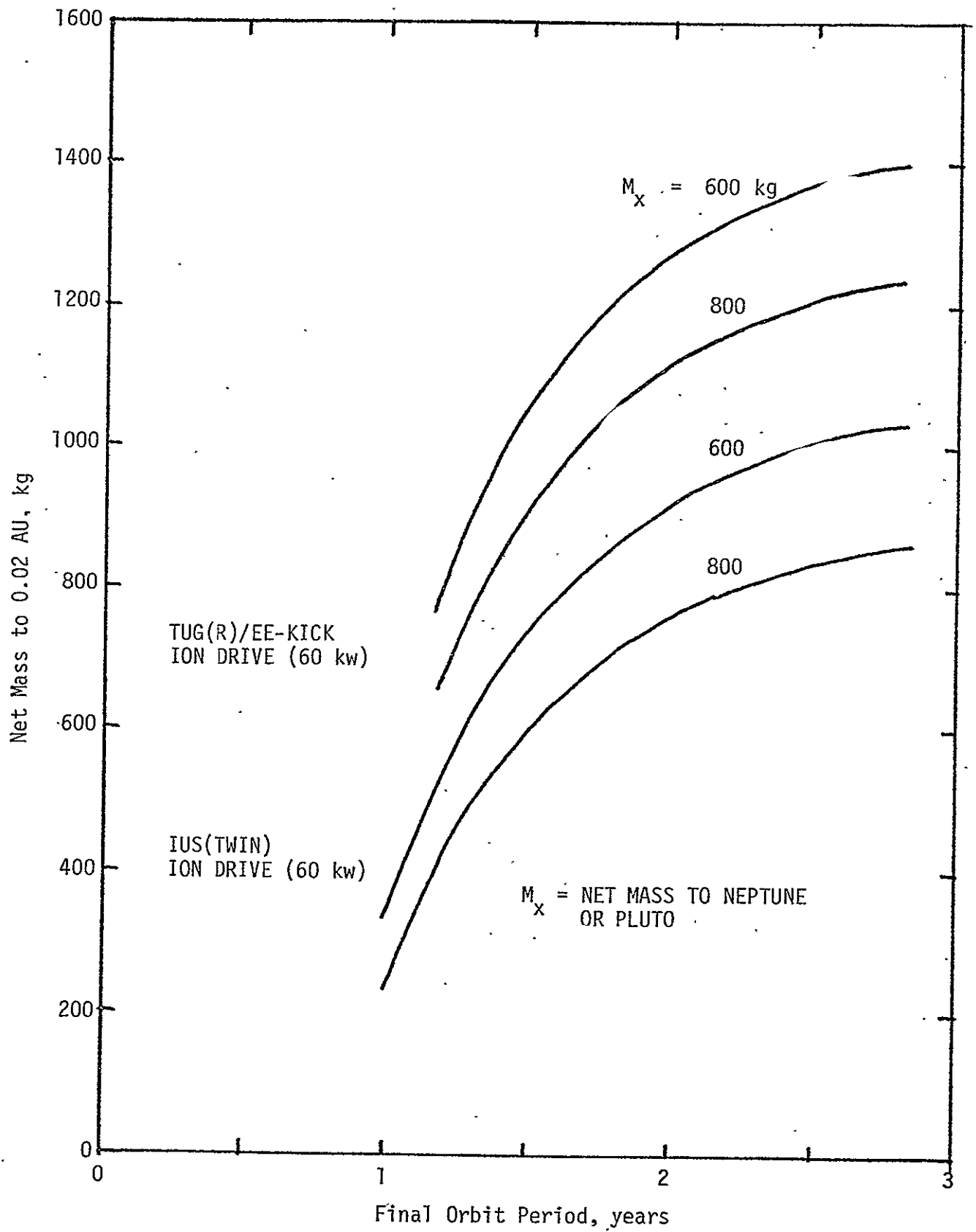perihelion.

Fig. 5.   PAYLOAD VERSUS ORBIT PERIOD FOR SOLAR PROBE MISSION

The smaller 30 kw low-thrust propulsion system is unable to deliver more than the two payloads to Jupiter on a direct trajectory with an appropriate arrival velocity. A good margin at Jupiter is necessary because it is the propellant used to reduce the orbit period of the solar probe. A possible solution is the use of a SEEGA (Solar Electric Earth Gravity Assist) trajectory which would add about 800 days to the flight time.

3.7 Atmospheric Entry Probes

Rationale

Atmospheric probes have been used or are planned for planetary studies at Venus, Jupiter, Saturn and Titan. In all cases, the major objective is to obtain a vertical profile of basic in situ data on the structure and composition of the atmospheres. The average properties of the Earth's atmosphere are well-known. Variations are studied using aircraft, balloons and sounding rockets. In the future, this detailed vertical structure information for both the Earth and the planets could be obtained with atmospheric probes. The proposed concept for using probes at Earth is based upon delivery of many probes to orbit as a partial Shuttle payload and recovery of all systems for subsequent reuse. This is necessary to make the approach cost competitive with other ways to obtain similar data. If such a concept can be developed, then a new technology base will have been established which will reduce the design and construction expenses for subsequent, more sophisticated planetary probes.

Science Instruments

All entry probe missions have some basic commonalities in science objectives and in science instruments. This is clearly seen in the payloads of three planetary entry probes (see Table 10). In all cases the basic instrumentation includes measurements of atmospheric structure (i.e., pressure, temperature, density and wind velocities), the transport

Table 10

ATMOSPHERIC ENTRY PROBE PAYLOADS

| Instrument | Pioneer Venus | | Galileo | Measurement Objectives |
	Large	Small		
Atmosphere Structure	2.3 kg	1.2 kg	3.0 kg	Pressure, Temperature, Density and Winds
Nephelometer	1.3	1.3	2.5	Cloud Structure
Cloud Particle Size Spectrometer	4.3	-	-	Number and Sizes of Particles
Infrared Radiometer	2.6	-	-	Thermal Flux
Neutral Mass Spectrometer	10.3	-	9.5	Composition
Gas Chromatograph	6.2	-	-	Composition
Solar Flux Radiometer	2.0	1.0	3.0	Energy Balance
He Interferometer	-	-	1.2	Composition
Lightning Detector	-	-	1.8	Lightning
TOTAL	29.1	3.5	21.0	

of energy through the atmosphere and the structure of clouds. The larger probes measure atmospheric composition, too. Each of these measurements could also be used in the Earth's atmosphere. There would be greater interest in the fine details rather than the basic properties measured by these initial planetary entry probes. However, many future planetary missions will also be interested in details.

Most instruments listed in Table 10 use basic and general methods to acquire the desired data. These methods work in atmospheres with various compositions and structures. With care in the design, calibration and operation of these instruments, the measurements can have the accuracy and precision desired for studies of the Earth's atmosphere. For any atmospheric probe, consideration must be given to more specific measurements, such as the He interferometer for the Jupiter atmosphere, and a combination of the gas chromatograph and mass spectrometer techniques so as to remove the inherent ambiguities in mass spectrometer data and to allow higher sensitivities to trace gases.

It is expected that a science payload of 20 to 30 kg would be sufficient for the routine measurements of the Earth's atmosphere that are proposed here. The ability to substitute or add other, perhaps heavier, more specialized instruments would be a desirable feature. A data rate of about 1.0 kbps should be sufficient--this is four times larger than that for the large Venus probe. Power consumption is 30 to 90 w. The wide variation is the result of two instruments with high power demands, the particle size spectrometer which uses 20 w and the gas chromatograph which needs 40 w.

A Reusable Atmospheric Entry Probe

Each atmospheric entry probe needs a similar set of subsystems to support the science instruments and to carry out other mission requirements. For electronic subsystems there are similarities in function, in environmental requirements and reliability which can be translated into identical designs for all applications. For some nonelectrical parts identical designs may not be desirable because there are potential cost and weight savings available if the design is solely for use in the Earth's atmosphere. The discussion that follows assumes that any new probe will be similar to the Pioneer Venus Large Probe [1,13] or to the Galileo Probe [1].

The electrical subsystems in any atmospheric entry probe must provide a source of power, time and sequence events, accept science and engineering data from the experiments and other subsystems, store data (during the blackout period) and transmit the data to an Earth-based receiver. The approaches which can be taken to satisfy the operational requirement for data transmission are indirect via a relay satellite direct to the Earth, or, in the case of a probe for the Earth's atmosphere the data can be stored on the probe and readout after recovery. With the advent of the tracking and data relay satellites at geosynchronous orbit, the first approach is easily implemented for Earth entry. In fact, this is probably the best option, because using a geostationary satellite as a relay provides unlimited time for data reception compared with a finite time for any other relay satellite. The choice of entry locations is not limited compared to direct transmission to an Earth-based receiver, and it is the option that has the most commonality with planetary missions.

Consequently, all requirements for electrical subsystems, including the interface with the bus, can be solved using identical approaches, if not identical hardware, for all applications. The environmental requirements on electrical systems are also common to all missions-- they must operate with high reliability at moderate pressures and temperatures and at the high levels of deceleration experienced during entry. Power is provided from batteries at 28 V, regulated as required. The command sequencer stores commands and times intervals between commands. It can be a simple computer which can also accept data, store it, and prepare it for input to the radio transmitter. Separate subsystems are used to turn the power on and off and to activate various pyrotechnic devices. The 10 w transmitter sends data as a modulated S-band radio signal from an omni-antenna. For planetary applications it may sometimes be advisable to accept a lower data rate rather than burden the probe with a more powerful transmitter or a sophisticated antenna. The transmitted signal must also be useful for range measurements. This requires a stable oscillator on the probe and/or a receiver

for a stable signal transmitted to the probe. While one may be adequate, it is recommended that the probe have both capabilities.

The mechanical subsystems must be designed to provide a safe entry and descent and a controlled environment inside the probe. The probe must be placed on a trajectory to the desired entry point and it must have the proper attitude for entry. For planetary missions, the main spacecraft performs these functions for the probe. This works well because the spacecraft can correct its course after separation. The terrestrial probes should have a separate propulsion capability to perform the deorbit maneuver. If the bus were used, each deployment requires two maneuvers and as a consequence a large propulsion system would be needed on the bus. With the retropropulsion on the probe it may be necessary to add a simple attitude control system to assure proper probe orientation during the burn and at entry.

The requirements for the heat shield/aeroshell are determined by the entry conditions. Entry from low Earth orbit is somewhat easier than most planetary missions except Jupiter where entry is very difficult. Thus it is possible to design an expendable heat shield that is also applicable to most planetary missions. Consideration should be given to a reusable heat shield for Earth entry only, based upon the technology developed for the Shuttle orbiter. If the aeroshell is retained, then there must be provisions for the acquisition of scientific data, such as removing some small sections of aeroshell to uncover the sensors. Another problem is the protection of the reusable heat shield after entry so that it is recovered in good condition.

For engineering reasons, an atmospheric probe usually has a pressure vessel. For Earth applications this structure can be simple and light, because it must support a pressure difference of only 100 kPa (1 atm). However, pressure differences for planetary applications are ten to a hundred times larger, so the pressure vessel is thicker and heavier. The same spherical shape can be used, so that things inside

the vessel could be the same. The parachute and the thermal control system are examples of components that are not expected to be common to all applications. The design of a parachute is determined by time allowed for descent to the desired altitude and by the atmosphere itself. The probe design is virtually independent of the parachute characteristics. The thermal control design depends upon the incident solar energy, internal power dissipation, and the heat loss or gain with the atmosphere. Hopefully thermal control concepts can be found which allow a common internal core for the probes. Methods which are consistent with this desire are varying the amount of insulation, using radioisotope heat sources and adding inert material (e.g., water) with a high heat capacity.

The probe for Earth applications is to be recovered and used again. It is assumed that the parachute is large enough to allow recovery in the air by a small plane or to permit a landing on the surface for pickup by a search party. While being prepared for a subsequent mission the probe would be thoroughly checked out. Some new materials are needed, particularly retropropulsion, pyrotechnics, maybe parachutes, seals and perhaps an aeroshell (see discussion above). The size of the probe should be similar to the Pioneer Venus Large Probe which has a 1.42 m aeroshell diameter and a 0.73 m diameter inside the pressure vessel. The mass, however, should be about 200 kg which is significantly less than the 312 kg mass of the Venus probe. Much of the reduction is expected to be due to a thinner pressure vessel.

The Earth-Orbital Probe Bus

The concept for a probe bus described here carries up to 12 probes and utilizes 10% of the Shuttle payload. Functionally, the bus packages the probes for launch, provides a means for placing new commands into the probes, keeps their batteries charged and releases the probes at the right time and in an appropriate orientation for their retromaneuver. The bus could be designed for recovery by the Shuttle after all probes have been released.

If the probes have a diameter of 1.4 m, then a circular array of six probes has a diameter of 4.4 m (including 0.1 m between probes) which just fills the Shuttle cargo bay. A second group of six probes can be added filling in much of the empty space between the first six (the noses of the second group point in the opposite direction; see Figure 6). This leaves a 1.4 m diameter core area which can be used to house the bus support subsystems. The length of the bus is estimated to be 1.5 m, or just 10% of the available length in the Shuttle cargo bay. The mass of the bus should be about 500 kg so that together with 12 probes the package would weigh 2900 kg or 10% of the Shuttle capacity.

A Program Proposal

A reasonable program might consist of about 10 launches per year, providing 120 entries. To carry out such a program requires something like three probe buses and 40 probes. Thus each probe would make about three flights per year, or about 20 flights during a projected seven-year lifetime. Significant cost reductions should be possible when producing 40 identical probes. The cost of a mission with 12 probes is estimated to be about $1.2M, consisting of the following:

Prorated cost of probes and bus	$600K
Recovery and preparation of probes and bus for flight	200
Shuttle launch	.300
Data acquisition and reduction	100

For this estimate to be accurate, the Shuttle cost must be at the $10./kg rate given to small demand packages, the probe production cost must be less than $1M/unit, and the number of full-time personnel involved in recovery, refurbishment and data handling should be about 50. Each of these assumptions seems reasonable but must be confirmed by further study.
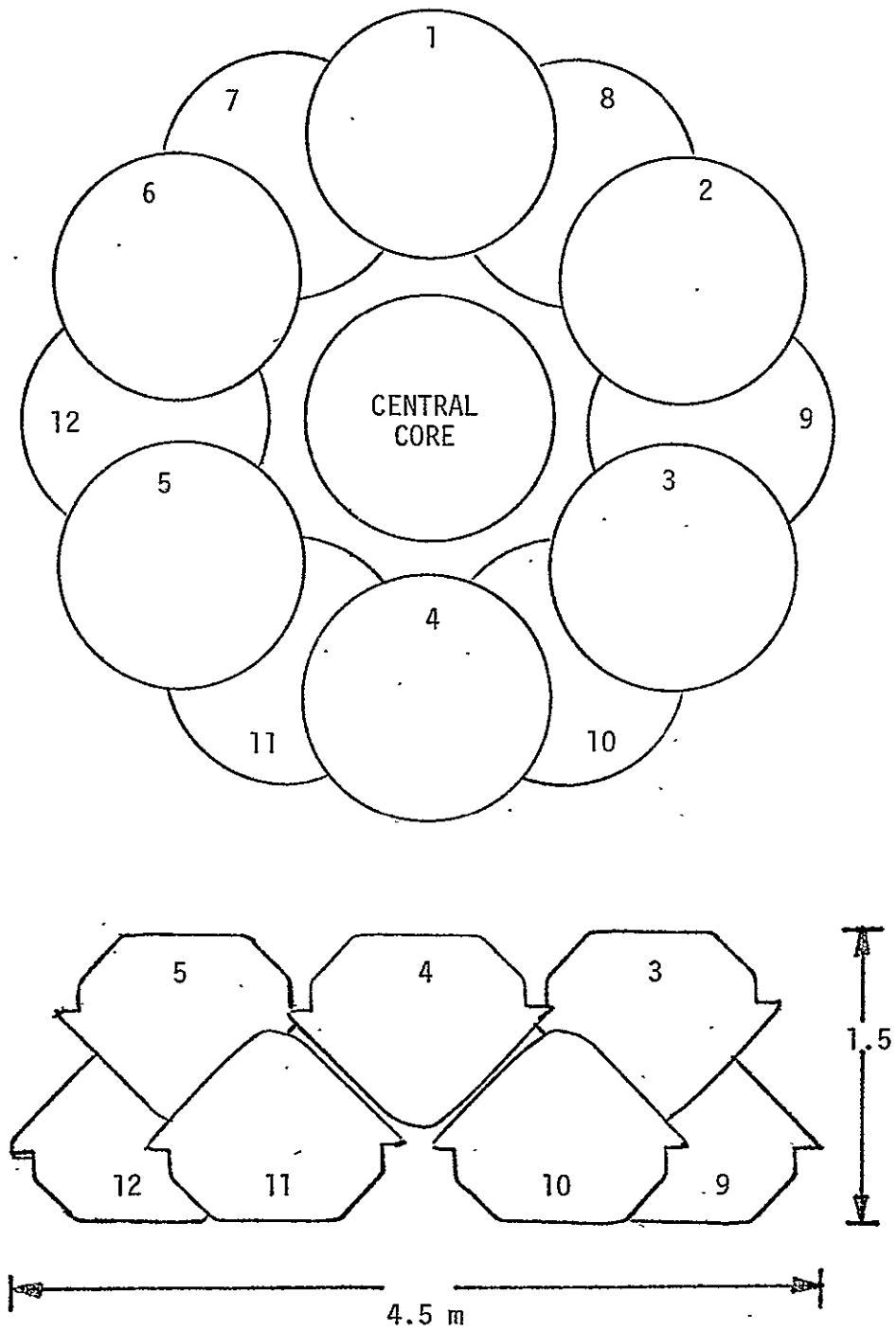
Fig. 6.   CONCEPT FOR AN ATMOSPHERIC PROBE BUS

3.8 Remotely Piloted Vehicles

Rationale

A remotely piloted vehicle (RPV) could be used in the exploration of a planet with an atmosphere, particularly Venus and Mars. The air-breathing RPV is now a well-developed concept for both military and civilian applications on the Earth. What is needed for common applications is an RPV which has a self-contained power source. The preferred system would be designed to operate at a pressure of about 5 mbar and can be used at an altitude of up to 40 km on the Earth or near the surface of Mars. At Mars, the RPV could be used to study the atmosphere, obtaining horizontal profiles of its properties, to look at the surface with high spatial resolution remote sensing instruments or to transport small payloads, such as surface samples or small experiment packages. Because of the low altitude and the low velocity, the RPV can provide high resolution coverage only of selected areas. Thus, an orbiter and an RPV are complementary. However, the RPV can acquire data over wider areas than a surface system (e.g., a rover) can. Such coverage would be valuable when planning surface mission operations or when interpreting scientific data from the surface.

Science Instruments

At Mars an RPV should be capable of carrying out several types of scientific missions. The first is a study of atmospheric composition and structure. The second is the investigation of the surface with emphasis on acquiring geomorphological, geochemical and geophysical data. The final possibility is transport missions such as taking small instrument packages for deployment at remote surface sites or bringing samples back to a central site for analysis or return to Earth. It is not known which mission the RPV will be performing since few studies of planetary missions have considered RPVs. Thus, payloads for each are discussed briefly; at the end of this section comparisons are made with payloads for terrestrial applications.

For atmospheric measurements the instruments shown in Table 10 are appropriate for RPVs as well as for atmospheric probes. A payload of about 25 kg is needed for missions dedicated to atmospheric objectives. The atmospheric structure and composition can depend upon such variables as latitude, time of day and season. Thus it is desirable to think of RPVs which have the capability of operating for two or more seasons, at any time and covering a wide range of latitudes. For the composition measurements, an uncontaminated sample is needed which is easily satisfied if the instruments are in the nose and the engine is to the rear. The optical instruments need both horizontal and vertical views. The RPV must provide power (up to 90 w) and a telemetry channel (up to 1 kbps) for this science payload.

Table 11 lists the candidate instruments for remote sensing of the surface of Mars from an RPV. Almost the same set has been proposed for an orbiter [1]. The RPV must be able to carry many, if not all, of these instruments. The RPV should be able to generate coverage in two ways. The first provides for complete coverage of major geological features whose area is up to 10^6 km^2. The second way is to make long flights flying over many geological units; this is similar to the coverage obtained with an orbiting spacecraft.

Table 11

CANDIDATE REMOTE SENSING INSTRUMENTS FOR MARS

Instrument	Weight	Power	Data Rate
Gamma Ray Spectrometer	13 kg	10 w	4 kbps
Reflectance Spectrometer	13	10	10
Imaging System	7	10	800
Radar Altimeter	10	20	1
Infrared Radiometer	7	.4	<1
Microwave Radiometer	15	12	<1
Magnetometer	2	2	<1
Gravity Gradiometer	20	20	<1

An RPV can operate from much lower altitudes than an orbiter does (e.g., 3 km instead of 300 km). This means that the spatial resolution can be finer. This is an important advantage for instruments like the gamma ray spectrometer, the magnetometer and the gravity gradiometer whose spatial resolution is roughly equal to the operational altitude. For some other instruments, particularly the radar altimeter, the microwave radiometer and the reflectance spectrometer, it may be possible to reduce the support requirements as a result of lower altitudes and/or less angular resolution. For the complete science payload, the estimated support requirements are a weight of 75 kg, power consumption of 75 w and a peak data rate of 800 kbps. The peak data rate assumes that one image is read out each 8 sec. However, the actual interval between scenes, which depends upon many system and operational parameters, could be much larger.

The transport of scientific packages is a third role for RPVs in Mars exploration. An experimental station, such as a penetrator or rough lander weighing up to 100 kg, could be taken by an RPV to a precisely known and well-documented location. These stations are used for long-term seismological and/or meteorological measurements or for ground truth data such as soil composition and images of surface material. Another possibility is the transport of surface samples from several automated rovers back to a single Mars ascent vehicle. These packages would probably be less than 10 kg. For either transport mission the RPV should be able to cover large distances, since remote station data or samples are desired from different geological units.

There are many terrestrial applications for RPVs. Most of the atmospheric and remote sensing techniques discussed for Mars are also potential techniques for studies of the Earth; however, the design of instruments would depend on planet-specific factors such as science objectives, operating altitude, etc. Consequently, an instrument designed for use at Mars may not be optimum for Earth application and vice versa. However, martian and terrestrial missions have commonalities

in terms of the support requirements for payload weight, power and data rate. On the basis of such similarities, a common airframe and perhaps power plant could be designed.

An RPV for Mars

Mars imposes some unique requirements and problems for an airplane. First, the atmosphere is thin; the surface pressure of 6 mbar is equivalent to an altitude of about 35 km on the Earth. Flight at such altitudes is possible but experience is limited. Second, the primary atmospheric constituent on Mars is carbon dioxide and there is no gas, such as oxygen, which could be used by an internal combustion engine. A third area of concern is control of the airplane: all flight activities must be executed automatically by on-board systems because human intervention is not possible. Each of these problems can be solved as shown by the flights of the NASA/FRC Mini-Sniffer [14] and other RPVs.

The airfoil design which works best at high altitudes is characterized by a high aspect ratio (>10), a high lift-to-drag ratio (>30), and a low wing loading (<10 kg/m^2). Similar features are found in gliders and in the Gossamer Condor. Scientific considerations favor a design optimized for cruise performance, specifically for long-range but not for speed of climbing. To maximize the payload, the airfoil and airframe design should make extensive use of modern, lightweight materials, such as composites and foams. Another factor which impacts the design is the fact that the airplane must be transported to Mars in a folded configuration and then unfurled after atmospheric entry.

A key element of a Mars airplane is the primary power source. As mentioned above, the atmosphere provides no useful gases for combustion. Consequently, a monopropellant engine concept is considered to be a good choice [14]. The preferred monopropellant is hydrazene (N_2H_4), a liquid which is easily converted to a high temperature, high pressure gas to operate a piston engine. Such an engine has been built and flight tested once. There is also interest in developing such an

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engine for Earth applications. Hydrazene could also be used as the propellant for small rockets providing a vertical takeoff and landing capability. A recent study [17] has shown that lithium batteries and an electric motor can be used to drive the propeller of a Mars airplane Such designs are expected to have almost twice the range of the air-plane powered by a hydrazene engine. Note, however, that the primary lithium battery cannot be recharged after it has been used.

The combination of a radioisotope heat source and a heat engine results in a concept for a power source that could keep an airplane up almost indefinitely. No other power source could equal the payload and endurance capabilities of the nuclear source. This option could be useful for terrestrial missions providing that the environmental impact of the nuclear power source is acceptable. Considerable devel-opment work needs to be done on the power plant and on a suitable air-frame to demonstrate that this concept will indeed fly. The airframe work can be done using a conventional gasoline engine whose weight and power output matches the anticipated values for the nuclear power source.

Performance estimates have been made for a battery-powered Mars airplane. From these studies the following are taken as typical char-acteristics [17]:

Payload	50 kg
Gross Weight	300 kg
Wingspan	21 m
Wing Area	20 m^2
Cruising Speed	300 km/hr
Endurance	30 hrs
Range	9000 km.

The range shown is for a one-way trip and it corresponds to 150° of latitude. Longer ranges can be achieved by carrying smaller payloads or by using higher initial weights. However, only one of the three missions discussed above--the transport of experimental packages or collected samples--is within the capabilities of this airplane. The other two demand an airplane with longer endurance, as would be the case if landings were permitted for refueling. Under these circumstances

the range is reduced to about 3000 km to account for the added equipment and fuel for takeoff and landing and for the necessity for round-trip flights. This round-trip range is sufficient for a single airplane to accomplish a survey of atmospheric structure over a wide latitude range. This range is also sufficient for geophysical studies of major geological features, and may be adequate for studies comparing many units. To carry out the many flights per airplane implied by these atmospheric and geological missions requires a large number of batteries brought from Earth. For a hydrazene powered airplane, the round-trip range is shorter, about 1500 km. To carry out these missions requires a large amount of propellant brought from the Earth or perhaps produced at Mars. In either case the mass which must be delivered to Mars is many times larger than the airplane's weight. Note, however, that either engine probably is adequate for most terrestrial applications, including flight testing related to the development of a Mars airplane.

The nuclear powered airplane differs significantly from the battery or hydrazene powered versions only in endurance and range, both of which are very long. The only limiting factor is the decay of the radioisotope heat source. The payload, gross weight, wingspan and area and the cruising speed should be similar to the values given above. At a speed of 300 km/hr, the plane can circumnavigate Mars in three days. In 40 days it can map a geological feature whose area is 10^6 km^2 with data taken in 3 km wide strips. Thus this type of airplane has the capability to do any of the missions described. It is the best choice for high scientific value and has the advantage of being able to do these missions without requiring additional support for refueling.

The Mars airplane needs electronic systems to control its flight and to interface with the science payload and with the outside world. It must use inertial sensors, altitude, air speed and wind data and the desired flight plans to determine the positions of control surfaces and to set the throttle. The airplane needs to know where it is. In the short term the inertial sensors can provide position information;

however, accurate position information must be provided periodically from a navigation beacon. Missions which must operate near the surface (this includes landings) require additional sensor capabilities, such as radar and image processing. The science instruments must have a system which provides sequenced commands to them and which accepts their output data. These data should be put into a format which accepts error detection and then used to modulate a transmitted radio signal. This signal would be received by a relay satellite for subsequent transmission to the Earth. All of the above electronic subsystems need power provided by a generator and/or a battery.

3.9 Fields, Particles and Gamma Ray Bursts

Particles and field observations have frequently been included in the scientific payloads of planetary missions. There is a continuing need for particle and field data at heliocentric positions other than that occupied by the Earth. Particle data are desired for the solar wind ions and electrons, the solar flare particles and the low energy cosmic rays. Field data consist of the magnetic field, the electric field and the electromagnetic waves generated by local plasma phenomena and by remote sources, especially the Sun. Extensive discussions of scientific objectives and anticipated results from such investigations can be found in References [15] and [16]. These observations can be made on almost any planetary or interplanetary mission, including the following which have been discussed in this report:

Mercury Orbiter
Mars Surface Sample Return
Neptune or Pluto Flyby
Synchronous Solar Observatory
Solar Polar Observator
Solar Probe.

Various instruments are available to perform these measurements. There are missions like Pioneer Venus with limited capabilities using three instruments weighing only 5 kg and also missions like Voyager capable of measuring all the above properties with six instruments

weighing almost 40 kg. The characteristics of the Voyager individual instruments are shown in Table 12; complete descriptions are in Reference [15]. It appears that planetary objectives (not shown in Table 12) account for some of the weight of these instruments. Thus, when planning future planetary missions, 10 to 25 kg of the science payload should, if possible, be allocated for particle and field instruments.

The locations of the recently discovered gamma ray bursts are determined by triangulation using time of arrival data. Two (or more) detectors on planetary spacecraft can be used to determine accurate source locations for identification with known astronomical objects. Such an instrument need not be large; the Pioneer Venus device is only 2.4 kg.

Table 12

VOYAGER PARTICLES AND FIELDS INSTRUMENTS

Instrument	Mass, kg	Power, w	Objectives	Capabilities
Plasma Particles	9.9	9.3	Density, energy spectra of solar wind ions and electrons	10 to 6000 ev; 50 to 1000 km/sec
Low Energy Charged Particles	7.5	4.1	Energy spectra, composition of nuclei, electrons	10 keV to 30 MeV/ nucleon; H to Fe
Cosmic Ray Particles	7.5	5.2	Energy spectra, composition of nuclei	1 to 500 MeV/nucleon; H to Fe
Magnetic Fields	5.5	2.2	Interplanetary magnetic fields	Three orthogonal components; 6×10^{-8} to 20 Gauss; dc to 8 Hz
Plasma Waves	1.4	1.4	Electric field intensity and frequency spectra	0.3 µV/m; 10 Hz to 56 kHz
Radio Astronomy	5.0	6.8	Solar radio bursts	20 to 1300 kHz; 2.3 to 40 MHz

4. CONCLUSIONS AND RECOMMENDATIONS

The Mercury Orbiter mission with solar/gravity science is the most favorable opportunity for accomplishing interesting planetary and multiple discipline science. This opportunity has good balance between planetary and other science objectives. The spacecraft technology or launch vehicle performance needed to do a Mercury mission appears to be sufficient for the expanded mission and within current capabilities. The combined mission will probably require some compromises in payload selection and to minimize other conflicts. Overall cost, however, should not be affected significantly.

The other single spacecraft opportunities also are favorable ones with respect to technology and/or performance requirements. For the following missions the primary emphasis will be on planetary objectives:

> Mars Sample Return with cruise sample
>
> Neptune or Pluto Flyby and solar system escape
>
> - Any planetary mission with interplanetary particles
>   and fields experiments.

In these cases the costs associated with the inclusion of the additional science should be minimal and justifiable by the added capability for scientific investigations.

The single launch vehicle opportunities are characterized by requirements for advanced propulsion systems (and/or SEEGA trajectories) and by a capability to do excellent planetary and solar observations. Because separate spacecraft are used, each of the two missions must be justified on its own merits. Consequently, these opportunities are not considered to be as favorable as the single spacecraft cases. The solar observatory at 0.2 AU and particularly the solar probe to 0.02 AU need additional study to determine that a spacecraft can be designed for these extreme thermal environments.

The two opportunities involving a common system require additional study to verify that there is a need for their development and to show that it is feasible to do so. In both cases; atmospheric entry probes for the Earth and RPVs for Mars, there are alternate ways to acquire similar science data. Furthermore, these studies may show that the applications at the planets and on the Earth have requirements that are not easily accomplished with a common system.

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