SMALL SATELLITE DESIGN FOR INNER SOLAR SYSTEM EXPLORATION

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This paper describes a study that examined using the Pegasus launch vehicle and small spacecraft for missions to the atmospheres of Venus and Mars. The launch vehicle can deliver approximately 820 lb (371.9 kg) into a 0° inclination circular orbit of 200 nm (370.4 km) altitude. With convential solid propellant rocket motors, a spacecraft of 73.8 kg (162.7 lb) would be delivered to Venus or Mars. A spacecraft using a currently existing 10 kW xenon-ion engine system with specific impulse of 3500 sec would deliver 143.4 kg (316.2 lb) to Venus or Mars. The spacecraft can fit inside the Pegasus fairing if folding solar array blankets are used.

INTRODUCTION

Most currently planned missions to Venus and Mars use large spacecraft and large expendable launch vehicles. For example, the Mars Observer spacecraft will be launched on a Titan III. Some of these missions could be accomplished with small spacecraft, thus requiring modest--and less expensive--launch vehicles.

MISSIONS

The Solar System Exploration Committee, an ad hoc committee of the NASA Advisory Council, was established to formulate planetary exploration missions through year 2000. Several missions to inner planets such as Magellan and Mars Observer were specified in the report. For this paper, the following missions were considered: Venus Atmospheric Probe, Mars Network Mission, and Mars Surface Probes [1]. The Venus mission is to place a Pioneer Venus-like probe in the atmosphere of the planet to examine chemical content. The Mars missions are for penetrators and surface stations to be parachuted to the surface of the planet to communicate with an orbiting spacecraft. The three missions are alike in that each uses atmospheric entry at the target planet for deceleration, so no large propulsion system is needed for orbit insertion.

PEGASUS LAUNCH VEHICLE

The Pegasus launch vehicle is currently being developed by Orbital Sciences Corporation and Hercules Aerospace Company. Pegasus can place approximately

820 lb (371.9 kg) into an equitorial circular orbit of 200 nautical miles (370.4 kilometers) [2]. This orbit was used as the initial orbit for all the mission studies. A weight of 820 lb is not a sufficient payload for a large spacecraft mission to Venus or Mars since the spacecraft must escape Earth orbit and achieve the proper heliocentric departure speed to reach the desired planet.

CHEMICAL PROPULSION

The current approach for interplanetary missions is to use a conventional solid propellant rocket motor to propel the spacecraft from Earth orbit into a trajectory to either Venus or Mars. For the following discussion, the orbits of the planets are assumed to be circular and coplanar. The speed of the Earth in its orbit about the Sun is 29.79 km/sec; the heliocentric departure speeds for trajectories to Venus and Mars are 27.28 and 32.73 km/sec, respectively [3]. Therefore, the velocity with respect to Earth upon escape v_∞ must be 2.51 km/sec (Venus) or 2.94 km/sec (Mars). The Δv to place the spacecraft on the desired heliocentric trajectory is [3]

$$\Delta v = \sqrt{v_\infty^2 + \frac{2\mu}{r_0}} - \sqrt{\frac{\mu}{r_0}}$$

where r_0 is the radius of the starting circular orbit. The value for r_0 is 6748.4 km, the mean radius of the Earth of 6378 km plus the altitude of 370.4 km. For Venus, the Δv is 3.469 km/sec; for Mars, it is 3.574 km/sec. The mass of fuel required to achieve this Δv can be found by using [4]

$$m_f = m_0(1 - e^{-\Delta v / I_s g_0})$$

where m_f is the mass of fuel required, m_0 is the initial mass of the spacecraft, I_s is the specific impulse in sec, and g_0 is the acceleration due to gravity at the surface of the Earth (9.81 m/sec^2). Most medium-sized solid propellant motors have specific impulse of approximately 285 sec with propellant mass fraction of 0.9 [4], so one-ninth of the fuel mass will be required for the case, nozzle, and other inert material. Using the above Δv values, the mass of propellant required is 264.4 kg for the Venus trajectory and 268.3 kg for Mars. The inert masses would be approximately 29.4 and 29.8 kg, respectively.

Table 1 presents a mass summary of a Pegasus-launched, solid propellant upper stage spacecraft for atmospheric missions to Venus or Mars. The rocket motor is sized for the Mars mission, so it will have enough propellant for the Venus mission as well. There are 73.8 kg (162.7 lb) of mass for the spacecraft. This must include any power, additional propulsion, and attitude control for the flight to Venus or Mars. For comparison, the Pioneer Venus small probes each had mass of about 91 kg (200 lb) and were about 61 cm (24 inches) in diameter [5]. Although the Pegasus-launched vehicle could not perform a Pioneer Venus small probe mission, the amount of mass is reasonable for many small satellite designs.

TABLE 1. Solid Propellant Design Masses

Solid Propellant	268.3 kg
Rocket motor inert mass	29.8 kg
Spacecraft	73.8 kg
TOTAL	371.9 kg

SOLAR ELECTRIC PROPULSION

An alternative approach to using conventional solid propellant upper stages is to use solar electric propulsion (SEP). Clearly, the inspiration for this paper comes from the Lunar GAS mission [6]. NASA has a currently existing xenon-ion engine system [7] that will be used in this study. The two-engine module draws a maximum input power of 10 kW to produce thrust of 0.4 N at 3500 sec specific impulse. The total mass for the module (excluding power-processing mass) is 70.2 kg (154.8 lb). The mass flow rate can be found by using the definition of specific impulse [4]

$$\dot{m} = \frac{T}{I_s g_0}$$

where T is the thrust and I_s is the specific impulse. The mass flow rate is 1.165×10^{-5} kg/sec for this system.

The article that details the xenon-ion engine module also describes a study for a lunar ferry that includes information about the large solar panels needed for solar-electric propulsion [7]. The panels for the ferry have specific mass at end of life of 6.0 kg/kW. The ferry would spend more time in the Van Allen belts than would an interplanetary probe since the ferry is slower and makes round trips. Ferry panels would sustain more radiation damage due to the increased exposure. The specific mass of 6.0 kg/kW will be used here for a conservative estimate of the solar panel mass. For 10 kW, a mass of 60 kg (132.3 lb) is needed. All thrusting takes place near the Earth, so the average solar flux value of 1353 W/m^2 is used.

The low-thrust trajectory can be divided into three components: Earth escape, acceleration to heliocentric departure speed, and coast to the target planet. Originally, the use of a lunar gravity assist during escape was considered, but eliminated because it would restrict launch windows due to the phasing of the Moon, planets, and spacecraft. The trajectory for Earth escape is an outward spiral from the initial orbit. Assuming that the orbit remains approximately circular, the time to escape is [8]

$$t_{esc} = \frac{1}{A}\sqrt{\frac{\mu}{a_0}}$$

where A is the acceleration of the vehicle and a_0 is the radius of the initial orbit. This is a conservative estimate (i.e., longer time) because the acceleration is assumed to be constant throughout the escape. In fact, the acceleration increases as fuel is depleted. With 0.4 N of thrust and an initial mass in low Earth orbit of 371.9 kg, the acceleration is 0.001076 m/sec^2. The initial radius of the orbit is 6748.4 km, so the time to escape is 7.143 x 10^6 sec (82.7 days). Multiplying this time by the mass flow rate yields the mass of fuel required for escape of 83.2 kg (183.5 lb).

Once the spacecraft has achieved escape, it is assumed to be in a heliocentric orbit at one AU from the sun with the orbital speed of the Earth. To achieve the necessary heliocentric departure speed for Venus or Mars trajectories, the ion engines must be fired to increase or decrease the spacecraft orbital speed. For this initial analysis, the firings were assumed to be short compared to the orbital period so as to be approximately impulsive. The initial mass m_0 is now 288.7 kg since 83.2 kg of fuel was used in Earth escape. Using the velocity increments of 2.51 km/sec (Venus) and 2.94 km/sec (Mars) in the equation for fuel mass yields requirements of 20.4 kg (Venus) and 23.7 kg (Mars). Dividing the larger (Mars) fuel mass by the mass flow rate gives a time of thrust of 2.034 x 10^6 sec (23.5 days). The transfer ellipse to Mars has a period of over a year, so the thrust time is reasonably short compared to the orbit period.

Table 2 gives a mass summary of the initial solar-electric propulsion design based on simplified trajectory analysis. Since the mission to Mars requires more fuel, it will be used for the common design. This SEP design has 134.8 kg (297.2 lb) for the actual spacecraft, an 82.7% increase over the chemically-propelled design. In addition, the SEP design has propulsion that can be used for course corrections and attitude control. The solar arrays have been overdesigned due to radiation damage, so there will be significant power available to the spacecraft until atmospheric entry at the target planet.

TABLE 2. Initial Solar Electric Propulsion Design Masses

Xenon-ion engine module	70.2 kg
10 kW EOL solar array	60.0 kg
Earth escape fuel	83.2 kg
Mars fuel	23.7 kg
Spacecraft	134.8 kg
TOTAL	371.9 kg

NUMERICAL SOLUTION OF LOW-THRUST TRAJECTORIES

The simple design techniques used to compute the low-thrust trajectories in the previous section are not sufficiently accurate for a final design, so a numerical solution was formulated. This formulation includes varying mass, so the fuel required to escape will be more accurately determined. For the Earth-Venus and Earth-Mars portions of the trajectories, the formulation includes effects due to the finite time of thrusting. The two equations of motion are [9]

$$\ddot{r} = \frac{F}{m} \cos \nu + r\dot{\phi}^2 - \frac{\mu}{r^2}$$

$$\ddot{\phi} = \frac{F}{mr} \sin \nu - \frac{2\dot{r}\dot{\phi}}{r}$$

and represent two coupled differential equations for planar motion using polar coordinates. The numerical solution was divided into two parts: Earth escape and interplanetary trajectory. The Earth escape portion starts from the low circular orbit with constant thrust (but increasing acceleration as fuel is depleted) in the direction of the velocity vector. The vehicle is considered to have escaped when its energy with respect to the Earth is zero. The interplanetary portion starts from the patched-conic heliocentric velocity, has a thrusting period to increase or decrease velocity, then a coasting period to the orbital distance from the sun of the target planet. The vehicle was targeted to impact the planet. These trajectories were not optimized, only calculated to determine whether or not the mission was feasible.

The equations were solved using a fourth-order Runge-Kutta numerical integration scheme implemented in True BASIC on an Apple Macintosh II microcomputer. The time to Earth escape was 5.888×10^6 sec (68.2 days), so the fuel required was 68.6 kg. The initial mass after escape is now 303.3 kg. For the acceleration to a trajectory to Mars, the spacecraft requires 16.8 kg and a thrusting period of 16.7 days. The fuel requirement for the Venus mission is higher since the spacecraft is thrusting closer to the sun. The fuel requirement for the trajectory to Venus is 29.7 kg with a thrusting period of 29.5 days. The updated masses (now using the Venus case since it requires more fuel) are in Table 3. There is an additional 8.6 kg (20.0 lb) for the spacecraft compared to the initial design that used simple trajectory analysis.

TABLE 3. Final Solar Electric Propulsion Design Masses

Xenon-ion engine module	70.2 kg
10 kW EOL solar array	60.0 kg
Earth escape fuel	68.6 kg
Venus fuel	29.7 kg
Spacecraft	143.4 kg
TOTAL	371.9 kg

SPACECRAFT DESIGN

The largest challenge for this design is fitting the spacecraft into the Pegasus fairing. The ion engines are 30 cm (about 12 inches) in diameter [10], so they can be placed in the fairing easily. The Lunar GAS mission stores 36 kg of xenon in a 0.01639 m^3 tank [11]. The Venus SEP mission requires 98.3 of xenon, so the tank volume must be 0.0448 m^3. This volume corresponds to a spherical tank 47 cm (about 19 inches) in diameter, so the tank can be placed inside the fairing easily. The spacecraft body is a rectangular box about 40x40x20 inches. Some of the ion engine components would go in the body. The payload would most likely be inside an aerodynamic shell for atmospheric entry, so the Pioneer Venus small probe design of about 20 inches diameter was used.

The previously quoted study of a lunar ferry [7] referred to an array power density of 200 W/m^2. Near the Earth where the solar flux is 1353 W/m^2, this array corresponds to 15% efficiency, a beginning of life value. To design for end of life, a factor of 1.5 will be used to take into account the radiation damage to solar cells. For 15 kW with 15% efficiency near the Earth, the array area must be 74 m^2. The spacecraft design has two solar panels, so each must be 37 m^2. The width of the panel is limited to 1.9 m by the fairing, so the length of each panel must be 19.5 m.

Figure 1 is a side view of the spacecraft in the fairing dynamic envelope. The arrays are blankets that can be folded. An interface structure connects the body to the launch vehicle. Figure 2 is a rear view of the spacecraft that shows the two ion engines and the folds of the array blankets to fit in the circular cross-section fairing dynamic envelope. Figure 3 is a side view of the spacecraft in orbit. The concept is for the two arrays to each have a single degree of freedom rotary joint to allow sun tracking as the body rotates in the orbit to keep the thrust vector aligned with the velocity vector. Only a small portion of the upper array is shown; the lower array is not shown.

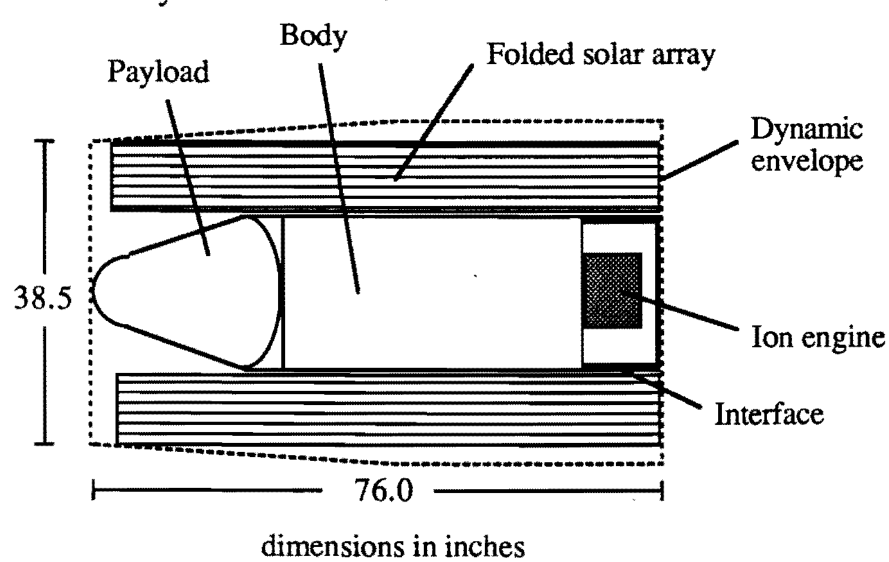

Figure 1. Side view of the spacecraft in the launch vehicle dynamic envelope

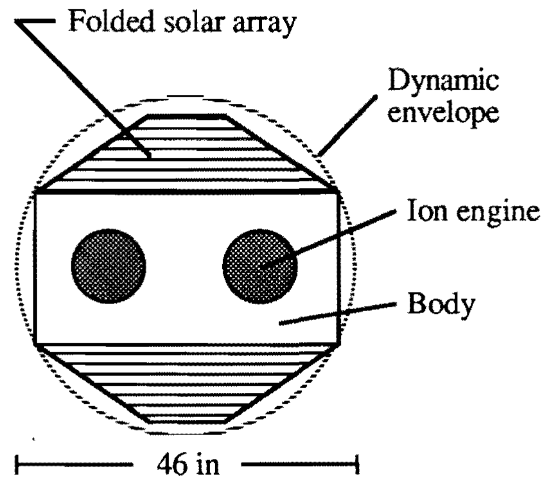

Figure 2. Rear view of the spacecraft in the launch vehicle dynamic envelope

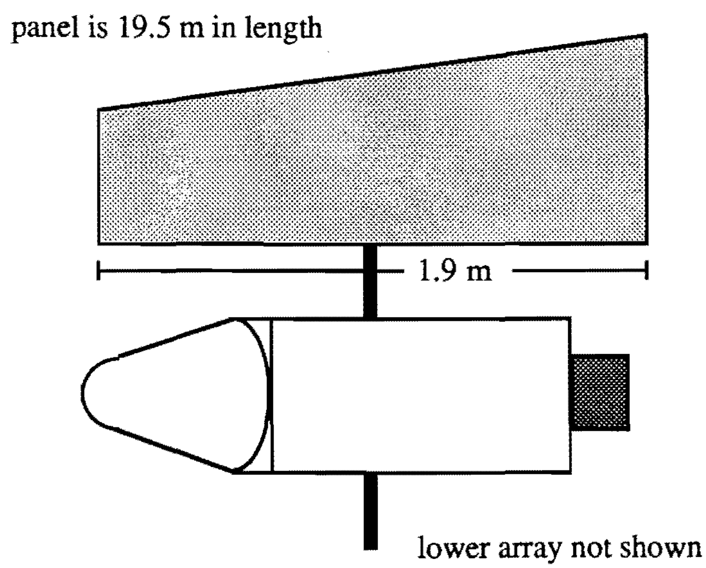

Figure 3. Side view of the spacecraft in orbit after array deployment

SUMMARY

A solar-electric propulsion spacecraft using the Pegasus launch vehicle can perform some currently planned inner solar system exploration missions. The combination of the small launch vehicle and SEP provides much mission flexibility because launch windows are no longer as critical. The cost of a small spacecraft will lower, so many spacecraft could be built for production line economy. This mission could also have the important task of being an inexpensive testbed for solar-electric propulsion.

REFERENCES

1. **Planetary Exploration through Year 2000: A Core Program**, National Aeronautics and Space Administration, Washington, DC, 1983, pp. 142-146.

2. **Pegasus Payload Users Guide**, Orbital Sciences Corporation, Fairfax, VA, 1989, pp. 23-24.

3. **Fundamentals of Astrodynamics**, Roger R. Bate, Donald D. Mueller, and Jerry E. White, Dover Publications, Inc., New York, NY, 1971, pp. 359-379.

4. **Spacecraft Systems Course Notes: Chapter 5. Spacecraft Propulsion**, James F. Smola, Applied Physics Laboratory, Laurel, MD, 1988.

5. **Pioneer Venus Case Study in Spacecraft Design**, R. F. Brodsky, American Institute for Aeronautics and Astronautics, Washington, DC, 1979.

6. "To the Moon on Gossamer Wings," Kerry T. Nock and Ronald P. Salazar, **Aerospace America**, March 1989, pp. 4C-42.

7. "Ferry to the Moon," Graeme Aston, **Aerospace America**, June 1987, pp. 30-32.

8. **Spaceflight Dynamics**, William E. Wiesel, McGraw-Hill, New York, NY, 1989, pp. 89-90.

9. **Ion Propulsion for Space Flight**, Ernst Stuhlinger, McGraw-Hill, New York, NY, 1964, pp. 118-124.

10. **The NASA Low Thrust Propulsion Program** (NASA Technical Memorandum 102065), James R. Stone and Gary L. Bennett, National Aeronautics and Space Administration, Washington, DC, 1989, p. 10.

11. "An Approach to the Design of a Simple, Radiation Hard, Highly Integrated and Reliable, Lightweight Spacecraft Bus," K. T. Nock, R. P. Salazar, C. Buck, and D. Eisenman, **AIAA/DARPA Meeting on Lightweight Satellite Systems**, American Institute of Aeronautics and Astronautics, Washington, DC, 1987, pp. 157-165.