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Conf-720104--6-Rev

BNL--46287-Rev.

DE92 010160

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Submitted to the
Ninth Symposium on Space Nuclear Power Systems
Albuquerque, NM
January 12-16, 1992

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HUMAN ROUND TRIP TO MARS: SIX MONTHS AND RADIATION-SAFE

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Abstract

We describe a different type of round trip to Mars, using a combination of spacecraft. Compared to typical proposals, this flight is relatively fast and relatively safe from biological radiation dosage. Our study is concerned with the trip from Earth orbit to Mars orbit. Four spacecraft are required for the round trip. The crew spends most of their time on board a comparatively large, well shielded spacecraft (LC) which is in free (non-powered) orbit about the sun. The crew travels from Earth orbit to the LC while on board a comparatively small, powered spacecraft (SC). At Mars, the procedure is reversed and the crew returns on a second LC. In addition, a cargo craft, with no crew, is sent to Mars prior to the crew leaving Earth orbit. The trip time is about six months and the radiation dose equivalent is within guidelines recommended by the National Commission on Radiation Protection and Measurements.

INTRODUCTION

The Radiation Environment

Outside the Earth's atmosphere, the radiation environment is quite severe. There are five sources of this radiation.

1. Galactic Cosmic Rays (GCR) are highly energetic particles which originate from outside the Solar System. They consist of 98% protons and heavier ions and 2% electrons and positrons. Their energies range from about 100 MeV to 10 GeV per nucleon. The radiation dose-equivalent (DE) from these particles is about .45 Sv/y (45 rem/y) to a human being with 2 g/cm² of Al shielding (NCRP Report No. 98, 1989).
2. Solar Event Particles (SEP) are energetic particles which originate from Solar Particle Events (SPE) of our Sun. They also are atomic nuclei, but their kinetic energies range from about one to 100 MeV per nucleon. The DE from these particles varies drastically according to the strength of the SPE. The frequency of occurrence of anomalously large SPEs ranges from several per year to one per decade.
3. Continuous solar cosmic rays (SCRs) are the relatively low energy particles given off by our Sun at a constant rate. Their energies range up to about two MeV per nucleon.

4. Trapped radiation consists primarily of protons and electrons confined to the "Van Allen Belts." They contribute a significant DE to astronauts only when a spacecraft passes through them.
5. If a nuclear engine is used, the radiation from the engine will be another source of DE. However, because of the distance from the crew to the engine (about 100 meters), and the large amount of structure and liquid hydrogen (LH) propellant between them, the radiation effects of the nuclear engine are expected to be small.

The National Council on Radiation Protection and Measurements (NCRP) has set up a guideline of .50 Sv (50 rem) for humans in outer space (NCRP Report No. 98, 1989).

Other Effects

There are two other effects which have quite serious implications for space travel. One is the psychological effect of loneliness during an extended (months) journey through space with very limited human interaction. The second effect is the physiological one of weightlessness for an extended period of time.

These factors, together with the danger inherent in the new technology of space travel, cause a voyage to Mars to be a dangerous enterprise. However, the adverse consequences of these effects will be reduced by a shortening of the trip time. This reduction is the main purpose of this study.

DESCRIPTION OF THE SYSTEM

Simplified Description

We propose an interplanetary transportation system which will provide a round trip for humans travelling from Earth to Mars and back. Compared to typical proposals, the journey will be relatively fast and relatively safe from radiation dosage.

The basic concept consists of four space vehicles and their trajectories. First, a heavily shielded, large spacecraft (LC#1), which usually is unmanned, is put into a free (non-powered) heliocentric orbit which closely passes Earth and Mars, in that order. Second, but during the same time frame, a second large spacecraft (LC#2), which will be used for the return trip, is put into an orbit which closely passes Mars and Earth in turn. Third, but also during the same time period, an unmanned cargo vessel, containing LH and other supplies, arrives at Mars before any humans depart from Earth. Fourth, a comparatively small spacecraft (SC) carrying the crew is launched from Earth orbit, travels to a rendezvous point with LC#1 and couples to it. The SC carries the minimal amount of supplies and only enough fuel to rendezvous with LC#1. Therefore, it is light and quickly can reach high speeds. The crew transfers to the relative comfort and safety of the LC. When they reach Mars, the SC decouples from LC#1 and orbits Mars, while LC#1 continues in its orbit. The crew spends 30 days on Mars, during which time the SC is re-supplied with materials from the cargo ship which was previously sent to Mars. The crew then leaves Mars in the SC and proceeds to a rendezvous point with LC#2 and couples to it. The crew transfers to it and travels back to the vicinity of the Earth. Then the crew transfers back to the SC and returns to the Earth orbit.

A More Detailed Description

Many options on this scenario are possible. However, to express it simply, we'll discuss one possible voyage, using a specific set of options. Then, we'll note some of the options.

Large Spacecraft #1 (LC1)

LC#1, without a crew, is put into a heliocentric orbit which will pass close to the Earth, and secondly, also will pass close to Mars (see Figure 1). The LC is powered by three Particle Bed Reactors (PBRs) (two nuclear electric reactors and one nuclear thermal reactor). It carries extra LH and most materials required by the crew. It is stocked by robots and its engines are automatic but can be overridden either by instructions from the crew (when they are on board) or by radio signals from mission control. It will be put into orbit over a period of months, or years, during which time, no humans will be on board. Propulsion will be supplied by very efficient but low thrust nuclear electric rocket engines.

The LC is heavily shielded and therefore does not have storm shelters. (A storm shelter is a small, heavily shielded volume in which the crew would be located in the event of an SPE.) The external shielding consists of the wall of the vessel, the surrounding store of supplies and the filled LH tanks. (Hydrogen is the most effective shielding per unit mass.) After the LC reaches Mars, the LH will be pumped to the SC, where it will be used as propellant.

Small Spacecraft (SC)

An SC, carrying the crew, is powered by two PBRs. This craft is envisioned as a scaled-down version of the Grumman Aerospace Corporation (GAC) Split/Sprint Mars Vehicle Design (Venetoklis 1991). That design is particularly good and has many laudable features.

Since the SC requires supplies and LH for a trip of about one day, it is relatively light, and therefore, relatively fast.

The SC is not as heavily shielded as the LC and therefore has a storm shelter. The crew will travel in the SC for about one day to intersect with the LC's orbit. The SC will couple to the LC and the crew will transfer to it and remain in it until the vicinity of Mars is reached. During the trip, the SC will be resupplied with cargo and LH from the LC. While at Mars, the SC will again be resupplied with cargo and LH, this time from the cargo vessel which previously was sent to Mars.

After spending 30 days on Mars, the crew, on board the SC, leaves the planet and meets up with LC#2. As before, the crew transfers to the LC, then returns to the vicinity of the Earth while resupplying the SC. Then, they again board the SC and return to Earth orbit. This is shown, conceptually, in Figure 2.

FIGURE 1. Orbits of the planets and LC=1.
Outbound trip (Earth to Mars)

FIGURE 2. Orbits of Earth, Mars, LC#1 and LC#2
Inbound trip (Mars to Earth)

Radiation

Of the five sources of radiation dose, we can neglect the effect of both the trapped radiation and the nuclear engine. The continuous SCR will be stopped by less than 1 mm of Al and, therefore, also need not be considered. This leaves the GCR and SPE radiation.

The NCRP has established a guideline of a dose-equivalent (DE) 0.50 Sv/y (50 rem/y) for astronauts. Therefore, for one year:

Total Allowable DE = DE (GCR) + DE (SPE), where DE (GCR) and DE (SPE) are the radiation DEs from the GCRs and the SPEs, respectively, in units of Sv.

The GCR DE is constant (in time) for a particular thickness of shielding. The SPE DE varies not only with the amount of shielding but varies dramatically with the intensity and duration of the event. As a standard, we'll use the anomalously large SPE of August 1972, which was the largest SPE recorded up to 1989. Using the data calculated by Letaw (1989), our reference shielding is 29 g/cm². This correspond to 8.9 cm of Al plus 5.0 g/cm² self-shielding for the blood forming organs. For this amount of shielding, we have:

.50 (total) = .30 (GCR) + .20 N (SPE), where N is the number of SPEs, each of which contribute as much DE as the August 1972 event. This equation has the solution: N = 1. That is, with 29 g/cm² of shielding, the astronauts can tolerate, indefinitely, the yearly equivalent of the August 1972 SPE. The LD required for one leg of the journey will provide about that much shielding.

Cycler

While the main objective of manned flights is to protect the passengers from radiation exposure with quick trips, that of unmanned cargo flights is economy. Low power, propellant expenditure, and lighter vehicles with reduced radiation shielding are important considerations. The operation would be very similar to that of a freight train following a regular schedule. Cargo units (CARS) would be dropped off into and picked up from local satellite orbits during flybys of Earth and Mars. This is essentially the operation of a cycler (Friedlander, 1986 and Aldrin, 1989).

Cycling transfer orbits have been proposed since the 1960s. The one we propose would follow the synodic period of Earth to Mars opposition. The vehicle would thus be in position to transfer cargo at each window opportunity. On Earth flyby the vehicle would ΔV into the next cycler orbit. We illustrate the actual orbits in Figure 3 using an idealized solar system for simplicity. We assumed the orbits for Mars and Earth to be circular and coplanar. The orbital parameters of an ellipse are:

1. Semi-major axis, a, in astronomical units (AU),
2. Eccentricity, e, and
3. Period, Y, in years. Their values are given in Table 1.

TABLE 1. Orbital Parameters

Orbit	Semi-Major Axis a(AU)	Eccentricity ϵ	Period Y (years)
Earth	1.00	0.0	1.000
Mars	1.52	0.0	1.874
Vehicle	1.60	0.393	2.025

The synodic period for outbound windows is 2.144 y. Note that the vehicle overlaps its orbit by 0.119 y to hit the window with Earth. At that point it ΔV s into the next cyclor orbit. The turn is 14.5° which rotates the vehicular orbital periapsis by 51.9° . In Figure 3 the elapsed times for the first three encounters with Earth and the first two encounters with Mars are given in years. The first Earth encounter is time 0.0. The trip time from Earth to Mars would be 0.399 y (146 d) and from Mars to Earth, 1.63 y (637 d).

The Earth flyby is used to rotate the orbit. The geocentric closing velocity (V_∞) is 6.5 km/s. The maximum turn angle is about 75° with 84.3° needed for the proper periapsis rotation. In the heliocentric system this corresponds to ΔV s of about 7.6 km/s out of 8.9 km/s needed. The additional 1.3+ km/s ΔV would require power. The exact value would depend on how close to Earth the vehicle passes. At Mars the vehicle would not execute a flyby and stay well away from the Martian surface. The V_∞ here is 9.8 km/s.

The total ΔV per trip is 1.3+ km/s at Earth flyby for the vehicle plus 6.5 km/s and 9.8 km/s at Earth and Mars respectively for local transfers with satellite orbits. This certainly qualifies for the low power and propellant expenditure desired. The low local relative velocities are due partially to low angle approaches to the planetary orbits. In the actual solar system allowing for the eccentricities and inclinations of the true planetary orbits along with other perturbations, the vehicle orbits would require additional adjustments to properly hit the windows. However, such a system still qualifies for low power/propellant transfer orbits for cargo shipments.

Though designed for cargo, a heavily shielded passenger "car" could be added for emergency passenger return if other vehicles failed. This would be a trip of 637 days unless a second cycling vehicle for the returning Mars to Earth window is used with the 146-day trip time.

Orbits

A computer program, ELLIPSE, was written to calculate parameters for, and information about, particular elliptical orbits. As input data for a particular orbit, the program requires the orbital period and the perihelion distance (the minimum distance to the focus of the ellipse, which is the Sun).

For ease of calculation, all orbits were assumed to be co-planar. Circles were used for the orbits of the planets and non-perturbed ellipses were used for the orbits of the spacecraft.

As a test case, the Hohmann Transfer was run and the results compared with those given in a standard astrodynamics textbook (Bates 1971). See Table 2.

TABLE 2. Comparison of Accepted Values for the Hohmann Transfer Parameters To Values Predicted By The ELLIPSE Model

Parameter	This Paper	Accepted Value ^a
Angle at Earth Departure (deg)	.11	0.00
Angle (θ) at Mars Arrival (deg)	179.83	180.00
Elapsed Time (d)	258.4	258.9
Speed of Earth (km/s)	29.79	29.78
Speed of Spacecraft (km/s)	32.73	32.73

^aFrom Bates 1971, p 365.

The angle, (θ), is the angle between the major axis of the orbit and a line connecting the Sun and the planet. See Figure 1.

For this paper, a reasonable orbit was chosen but not optimized. For this orbit, the period is 5.0 y and the perihelion distance is .72 AU (the orbital radius of Venus). The following results were obtained:

Parameters calculated for an Earth to Mars transfer with a period of 5.0 y and a perihelion of .72 AU:

Angle (θ) at Earth Departure	69.21 deg
Angle (θ) at Mars Arrival	102.93 deg
Elapsed Time	46.22 d
ΔV	$18.99 + 19.53 = 38.52 \text{ km/s}$

The outbound trip time (elapsed time from Earth to Mars) is quite reasonable at 46.2 d, without optimization. For a smaller perihelion, the trip time would be less but the LC (although uninhabited except between Earth and Mars) would pass closer to the Sun, thereby increasing its residual radioactivity.

For the return (inbound) trip, we calculate a requirement of 127 d (see Figure 2).

The total trip time is:

$$\text{Total time} = A + B + C + D + E + F + G$$

where A is the SC trip from Earth orbit to LC#1, accelerating only (1 d).

B is the LC#1 trip from Earth locale to Mars locale (46 d).

CYCLIC VEHICLE

FIGURE 3. First Two Orbits of Cyclic Transfer Vehicle. Points are Elapsed Time in Years Starting with in Earth Encounter at 0.0.

C is the SC trip from LC#1 to Mars, braking only (1 d).

D is the 30 day stay on Mars (30 d).

E is the SC trip from Mars orbit to LC#2, accelerating only (1 d).

F is the LC#2 trip from Mars locale to Earth locale (127 d).

G is the SC trip from LC#2 to Earth orbit, braking only (1 d).

Outbound time	= A + B + C	= 1 + 46 + 1	= 48 d
Stay on Mars	= D		= 30 d
Inbound time	= E + F + G		= 129 d

Total time	= 207 d.	<u>207 d</u>
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This trip was not optimized. The orbits of Earth and Mars were assumed to be circles. In reality, Mars' orbit is significantly elliptical and the minimum distance between the two planets is less than that between two circles having the same center. The calculated travel time is 177 d (48 + 129 d). If the trip is optimized, we estimate that this can be reduced by about 15%, to a value of about 153 d. The optimized total trip time is then 183 d (153 + 30 d), or about six months.

Options

An overriding principle in choosing any of various options is safety. Redundancy is always desirable for system elements so that a failed component can be bypassed and its function performed by another component. The balancing traits of redundancy are system complexity and cost.

Using this principle, the LC is designed with three engines, with their capability such that the mission can be accomplished with any two of them. Similarly, the SC is designed with two engines and its mission can be performed with either one alone.

An even more desirable option is to use two SCs, each with a complete crew, and each able to hold both crews (in case of the failure of one SC) for the limited required time.

DISCUSSION

Complexity. The LC/SC System is complex in that two rendezvous of the spacecraft are required. However, similar capabilities must be developed for any large scale space program - with or without human crews.

Funding. The LC/SC System also requires a large amount of funding. However, some of that funding can be amortized by using the LCs for transporting cargo with the Cyclor concept.

In general, a manned round trip to Mars is an enormous undertaking and must be sufficiently funded. Several spacecraft and several crews should be used. Even

if a continuous program of spaceflight to Mars is not envisioned; a single craft is not a viable option. It would be better to not attempt the journey than to undertake it with insufficient support.

CONCLUSIONS

The concept of an LC/SC transportation system between Earth and Mars is feasible and results in:

1. A tolerable, long-term radiation DE absorbed by the crew.
2. A relatively fast round trip to Mars for the human crew.
3. A continuous, cyclical program of moving cargo to and from Mars economically.
4. A possible method of transportation to the outer planets.

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Acknowledgments

One of the authors (O.W.L.) is grateful for discussions with Harold C. Berry and Arthur D. S. Harris of Brookhaven National Laboratory, New York. Also, this study is the result of concerns of trip time expressed by Franklin Chang-Diaz of NASA-Houston.

Work performed under the auspices of the U.S. Department of Energy under Contract No. DE-AC02-76CH00012.

Glossary

AU	Astronomical unit (the distance from the Sun to Earth)
CARS	Cargo units of the cyclor
DE	Dose-equivalent
GAC	Grumman Aerospace Corporation
GCR	Galactic cosmic rays
LC	Large spacecraft
LH	Liquid hydrogen
NCRP	National Council on Radiation Protection and Measurements
PBR	Particle bed reactor
SC	Small spacecraft
SCR	Solar Cosmic Rays
SEP	Solar event particles (which originate from Solar Particle Events [SPEs])
SPE	Solar particle events

Definitions

1. Radioactivity is the disintegration of nuclei together with the emission of particles of radiation. The S.I. unit of the rate of radioactive decay is the Becquerel (or Curie in atomic units).
2. Absorbed dose is the radiation energy absorbed per unit mass of absorber material. The unit is Gray (or Rad).
3. Dose-equivalent (DE) is the product of the quality factor and the dose. The quality factor is a measure of the effects of different types of radiation. The DE, therefore, is a measure of the biological damage caused by various sources of radioactivity. The unit is Sievert (or rem).

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