IMPACT OF LUNAR OXYGEN PRODUCTION ON DIRECT MANNED MARS MISSIONS

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A manned Mars program made up of six missions is evaluated to determine the impact of using lunar liquid oxygen (LOX) as a propellant. Two departure and return nodes, low Earth orbit and low lunar orbit, are considered, as well as two return vehicle configurations, a full 70,000-kg vehicle and a 6800-kg capsule. The cost of lunar LOX delivered to orbit is expressed as a ratio of Earth launch cost.

LIST OF ACRONYMS

TLI	Trans Lunar Injection
LOI	Lunar orbit insertion
TMI	Trans-Mars injection
MOI	Mars orbit insertion
TEI	Trans-Earth injection
EOI	Earth orbit insertion
EEC	Earth departure and Earth return capsule vehicle
EEF	Earth departure and Earth return full vehicle
LEC	Lunar departure and Earth return capsule vehicle
LEF	Lunar departure and Earth return full vehicle
LLC	Lunar departure and lunar return capsule vehicle
LLF	Lunar departure and lunar return full vehicle
ECCV	Earth crew capture vehicle
MTV	Mars transfer vehicle
ETV	Earth transfer vehicle
LEO	Low Earth orbit
LOX	Liquid oxygen
LH_2	Liquid hydrogen
LLO	Low lunar orbit

INTRODUCTION

As part of the SRS Manned Mars Mission and Program Analysis Study conducted for Marshall Space Flight Center, a series of manned Mars programs was investigated, each culminating with a manned base on Mars. Initial guidelines and assumptions for the study specify the 1999-2035 timeframe and that each program would include an exploratory, outpost, and base phase. The first three manned missions of each program were considered site survey missions. The outpost phase was reached when a portion of the crew had the option of remaining at Mars between missions. The base phase was accomplished when there was a permanently manned facility on the surface of Mars.

As Fig. 1 shows, each of the five example programs emphasized different objectives. The Early (or Reference) Program takes advantage of the cyclic nature of Earth-Mars oppositions (see Fig. 2) to accomplish the shorter exploratory missions near the oppositions when Mars is at perihelion, and represents a fairly vigorous program to reach milestones as soon as possible. The Phobos/Deimos Program has initial landings on the surface of the martian moons and later excursions to the surface of Mars. The Split/Sprint Program uses split missions with cargo vehicles sent on long, low-energy trajectories, and the lighter manned modules sent on shorter, high-energy trajectories. The Mid-Range Program delays the start of a manned Mars program while additional emphasis is placed on low Earth orbit (LEO) activities. The Later Program delays the missions for one cycle of oppositions, approximately 15 years, and includes a lunar emphasis program for the years 1999-2018. This paper investigates the possible benefits of a lunar liquid oxygen (LOX) propellant plant on the Later Manned Mars Program.

Studies on the impact of lunar LOX production on Mars missions have been conducted by several authors (*Keaton*, 1985; *Babb and Stump*, 1986; *Cordell and Wagner*, 1987). These studies have all been based on conjunction class "minimum energy" missions. Conjunction missions use low-energy near-Hohmann transfer trajectories, which cause flight times to Mars to be on the order of eight months each way. Waiting at Mars for a near-Hohmann return opportunity requires stay times at Mars of 300 to 550 days. Thus the total mission time for a conjunction class mission is 2.5 to 3 years. Conjunction class missions provide the most efficient use of propellants when transporting large amounts of material to and from Mars. Sensitivity studies (*Davis*, 1986) have shown that the conjunction class mission can deliver four times the payload on a round-trip mission or six times the one-way payload from Earth orbit to Mars orbit than the opposition class mission (with the same initial mass in LEO and propulsion technology). The disadvantage of the conjunction class mission lies in the long total trip time that can be debilitating on the crew and increase propellant boil-off losses. For this reason the opposition class mission is important in planning Mars programs. The opposition class trajectory is characterized by 30-90-day stay times at Mars and total trip times from 450 to 730 days. This paper considers opposition class missions for the manned exploratory and early outpost phases of the program and

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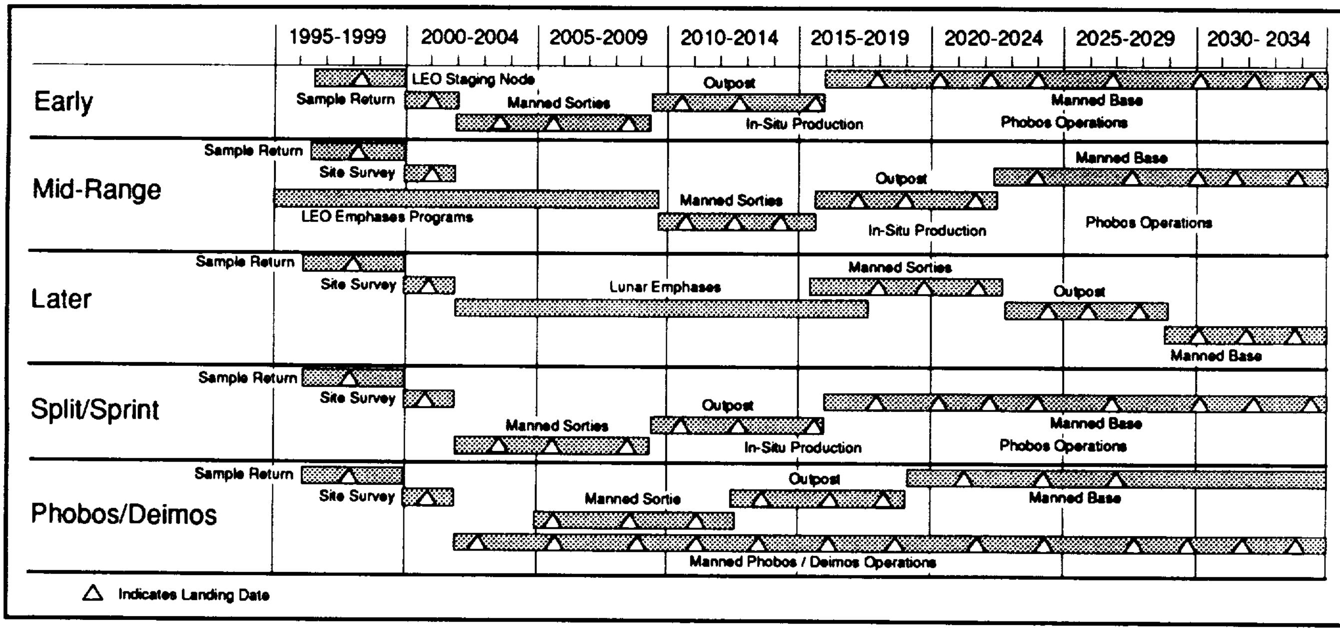

Fig. 1. Example of the manned Mars mission programs.

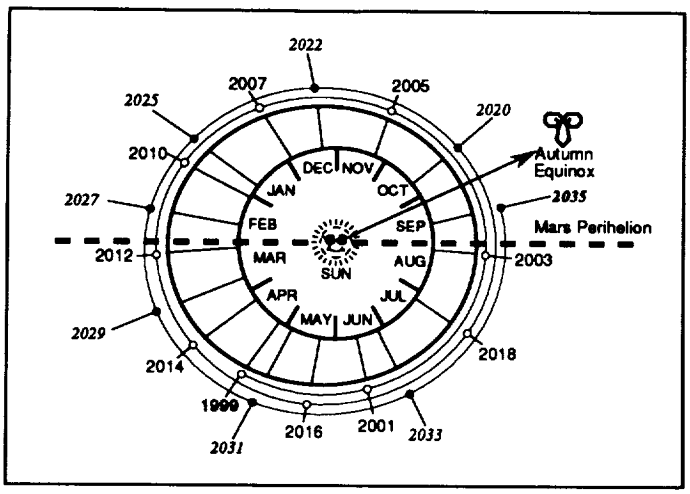

Fig. 2. Earth-Mars oppositions.

conjunction class missions for the later outpost and base phases of the program. Table 1 details the missions required to accomplish the program with the payloads carried on each mission.

COMPARATIVE SCENARIOS

The LOX propellant required for the various missions in the Later Program was assumed to be produced on either the Earth or the Moon. In either case a 500-km circular space station orbit is used as a staging area for assembly of the Mars transfer vehicle (MTV). If the LOX is produced on the Earth the trans-Mars injection (TMI) burn is initiated from the LEO staging orbit. If the LOX is produced at a lunar propellant plant the MTV is loaded at the staging area with all the liquid hydrogen (LH_2) required for the mission, including the lunar operations, and with enough LOX to initiate the translunar injection (TLI) and lunar orbit insertion (LOI) maneuvers. The ΔVs for these burns are 3.155 km/sec and 0.975 km/sec respectively (*Babb and Stump,* 1986). The resulting lunar orbit is a 500-km circular orbit. This study assumes that the lunar LOX is available in the same orbit. The MTV is then

TABLE 1. Mission characteristics for the "Later" Program.

Mission Number	Mission Class	Opposition Year	Trip Time in Days	Stay Time in Days	Payloads
1	Manned opposition	2018	450	60	130,000 kg
2	Manned opposition	2020	550	60	130,000 kg
3	Manned opposition	2022	730	90	130,000 kg
4	Manned opposition	2025	730	90	178,000 kg
5a	Manned opposition	2027	730	90	130,000 kg
5b	Cargo conjunction	2026	220	NA	100,000 kg
6	Manned conjunction	2029	1000	500	230,000 kg

tanked up with the lunar LOX and the TMI operation is initiated. The optimum time for lunar departure is at a full Moon when the velocity vectors for the Earth and Moon are in the same direction. All departures from and returns to the Moon were therefore constrained to occur at the time of the nearest full Moon.

The MTV used in this analysis is a three-stage system with each stage's propulsion system having an I_{sp} of 482 sec. The propellant used is a mixture of LOX and LH_2 with a mixture ratio of 6:1. The first stage performs the TMI burn and is then expended. The second stage performs a 25 m/sec midcourse correction, the Mars orbital insertion (MOI), and trans-Earth injection (TEI) maneuvers. The MOI maneuver is accomplished with an aerobrake provided the C3 at Mars is less then 50 km/sec, otherwise a propulsive burn is required to reduce the C3 to 50 km/sec. A 24-hr 500-km periapsis highly elliptical parking orbit around Mars was assumed to accommodate Mars landing operations and minimize departure ΔV requirements. After the appropriate stay time at Mars, the second stage is ignited to perform the TEI. Following this burn the second stage is expended and the Earth transfer vehicle (ETV) utilizes the third stage for a midcourse burn of 25 m/sec.

At this point in the analysis two return vehicle configurations are investigated. A full module return brings back the entire ETV with a mass of 70,000 kg. A capsule return expends the ETV and crew return is completed in a small Apollo-type Earth crew capture vehicle (ECCV) with a mass of only 6800 kg. Two return nodes are also investigated in the study: return to a 500-km circular low lunar orbit (LLO) or return to a 24-hr 500-km perigee highly elliptical Earth orbit. If the ETV or ECCV is returned to Earth orbit, an aerobrake is assumed for the Earth orbit insertion (EOI) maneuver if the C3 at Earth arrival is less then 25 km/sec. If the C3 is greater than 25 km/sec the EOI maneuver is performed propulsively by the third stage. In both the full return and capsule return to the lunar orbit, the LOI is performed by the third stage. As was the case with lunar departure, return to the Moon is accomplished at a full Moon to minimize the ΔV requirements. Figure 3 depicts the various options available for departure and return nodes.

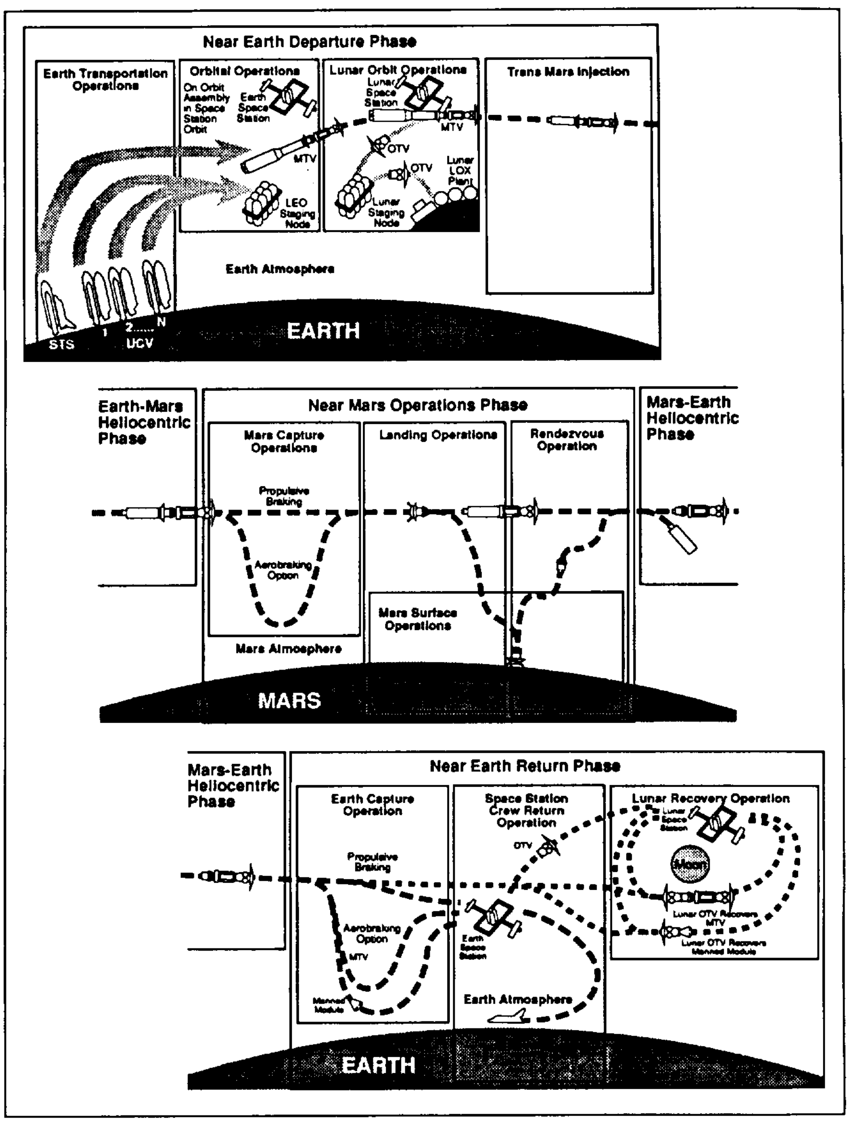

Fig. 3. Options available for departure, arrival, and return nodes.

RESULTS AND CONCLUSIONS

The total trip time, stay time at Mars, and payload masses required for each mission are inputs to the SRS trajectory analysis code. Using Lambert's theorem to calculate ΔV requirements, the program determines the trajectory, fuel requirements, and vehicle size to minimize the mass in LEO or LLO, depending on departure point. If a lunar departure is planned, the fuel requirements for the TLI and LOI are also calculated and charged to the mass required at the LEO staging orbit. The results of these calculations are presented in Fig. 4. Note that the mass of the payload is not included in the total mass since it is constant for each mission. As can be seen from Fig. 4 there is a wide variation in the total mass required to accomplish each mission in the Later Program. This is due to the different payloads carried (see Table 1) and the oppositions variations in Earth/Mars (see Fig. 2). As would be expected, the capsule returns are the best options due to the much smaller mass of the ECCV. In addition, the initial mass required in LEO for lunar departure, including the extra propellants for TLI and LOI, are generally less than the mass in LEO required for direct missions from the Earth to Mars. The lunar departure-Earth return options provide the best performance since the ΔV requirements for TMI are smaller when departing from lunar orbit and the ΔV requirements at Earth return are smaller due to the higher orbital velocity in LEO. The lunar departure-lunar return options are generally the worst, due

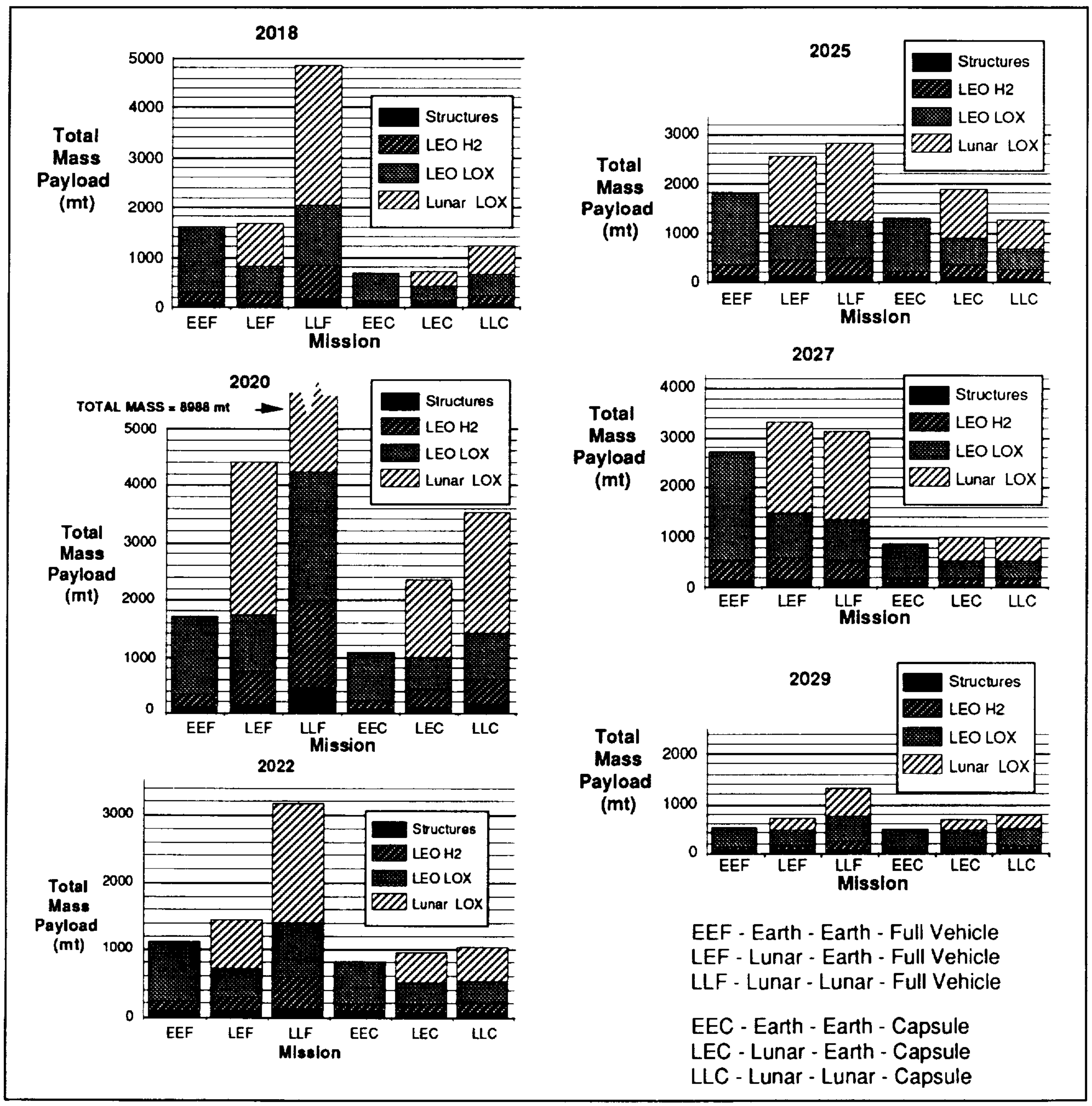

Fig. 4. Mass requirements for missions in the "Later" Program (1 mt = 1000 kg).

primarily to the large ΔV maneuver required to place the vehicle in LLO upon return. Figure 5 shows ΔV requirements for the different missions and options.

Determining the cost savings incurred by using lunar LOX is a difficult task. The total cost of LOX in LEO is dominated by the Earth surface to LEO transportation cost, with manufacturing cost being negligible. The transportation cost of lunar LOX should be less, based on the smaller gravitational field; however, the manufacturing cost should be higher due to the harsh lunar environment. This paper has assumed that LOX is present in both LEO and lunar orbit without looking at the manufacturing and transportation cost. But a relative cost ratio of lunar LOX to LEO LOX can be assigned, as shown in Fig. 6. For example, if you save 400,000 kg of total propellant in LEO by going to the Moon and picking up 800,000 kg of lunar LOX, then the relative cost ratio of the lunar LOX is 0.5. In cases where the additional LOX and H_2 required in LEO to push the MTV to lunar orbit is greater than the LEO savings incurred by the use of lunar LOX, it is not clear there is a savings and these missions are assigned a value of 0 in Fig. 6. The results to date are inconclusive as to the viability of lunar LOX for use in direct manned Mars missions. Additional study in the areas of propellant production on the Moon and lunar surface to orbit transportation systems must be conducted before it can be determined if relative cost ratios such as those shown in Fig. 6 are technologically feasible. As the definition of the lunar base progresses and Mars mission planning matures, lunar LOX propellant production should be analyzed in the framework of total program life cycle cost.

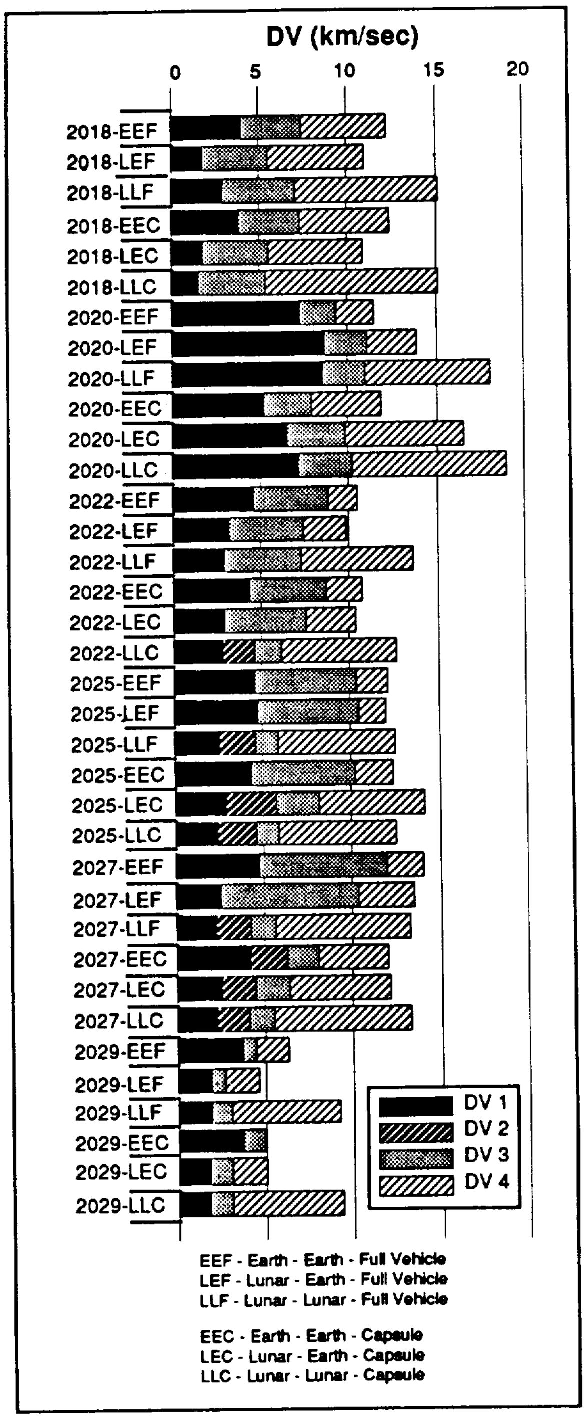

Fig. 5. Delta V requirements for "Later" Program options.

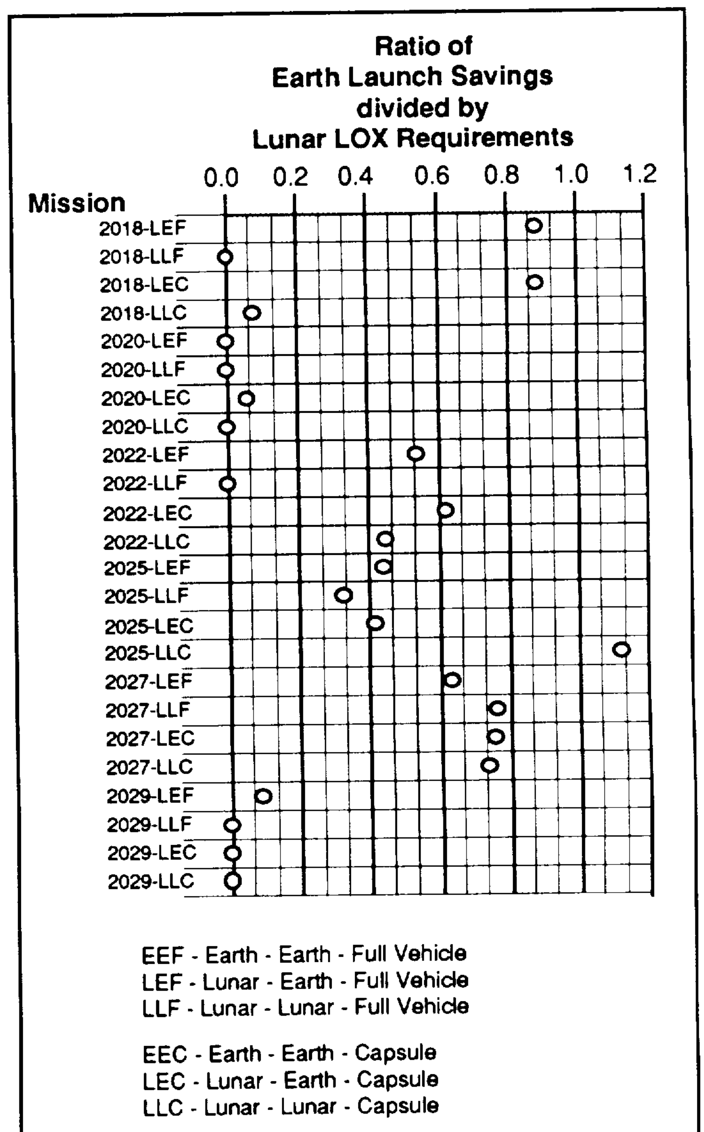

Fig. 6. Lunar launch cost relative to Earth launch cost.

NOTE IN PROOF

This paper presents results only for "direct" manned Mars missions, i.e., those without gravity assists and without considering libration point basing options. Several papers have been published since this paper was presented that give insight into benefits to be gained from both gravity assist and from libration point basing options. These publications are now included in the bibliography of this paper. All these papers are somewhat qualitative in nature, and it is evident that more refined total life cycle cost analyses must be accomplished before one basing option can be designated as optimum for performing manned Mars missions.

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