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EARTH VICINITY TRADES AND OPTIONS

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ABSTRACT

The options for recovering a returned manned Mars spacecraft are surveyed. Earth parking orbits from libration point to low circular are discussed, with a 500 km perigee, 24 hour period elliptical orbit chosen as a baseline for further calculation. Several techniques for recovering up to 100 metric tons of returned spacecraft are investigated, including recovery by a LEO based OTV pushing the spacecraft to LEO, an OTV transporting an aerobrake to the spacecraft, and an OTV delivering propellant to the spacecraft. Methods utilizing OTVs result in less total mass in LEO, but may not be the minimum cost solutions if significant development and testing are required.

INTRODUCTION

A number of methods exist for recovering a manned Mars mission crew and spacecraft in or near Earth orbit. The parking orbit, mass, and volume of the returned spacecraft must first be determined, then a technique can be chosen to return this mass to low Earth orbit (LEO) for refurbishment.

PARKING ORBITS

Options for Earth parking orbits on return of a manned Mars mission range from high circular, perhaps including a libration point and high elliptical; with periods on the order of 48 hours, to low apogee elliptical and low circular; or direct entry into the Earth's atmosphere. All these options, with the exception of the last, assume propulsive insertion.

The high circular parking orbits are most appropriate for electric propulsion stages. References 1 and 2 discuss these mission scenarios. If multimegawatt power supplies are available, electric propulsion may prove to be attractive. It is a special case, apart from high thrust propulsion, however.

Electric propulsion trajectories consist of many-revolution spirals, due to the low, usually continuous thrust levels, and are thus constrained for all practical purposes to circular orbits. A manned electric propulsion stage cannot spiral up or down through the radiation belts with a crew aboard because of the many months required and high radiation dose involved. Also, radiation-sensitive equipment (including integrated circuits sensitive to logic level upsets, etc.) may not be able to stand such radiation levels unless protective shielding is provided. A high-thrust boost through the belts is possible, but much of the performance advantage of electric propulsion may be negated. The high thrust delta V to geosynchronous orbit (4.2 km/sec, 3.82 with no plane change) is more than a typical trans-Mars insertion burn from the Space Station orbit for a conjunction class trajectory (3.8 km/sec). The electric propulsion stage must therefore either spiral up through the belts unmanned or be based beyond them. In either case, the crew must be brought up and retrieved from the interplanetary spacecraft parked in high circular orbit.

The altitude of this high circular orbit requires some study. Geosynchronous orbit (GEO) is a candidate. The 42 metric ton propellant capacity Orbital Transfer Vehicle (OTV) described later in this paper (Figure 5) can carry a 6 metric ton crew module round trip from the Space Station orbit to GEO and back.

The L2 libration point (the one behind the Moon, see Ref. 3) and low lunar orbit, have also been proposed as staging points for repeated Mars missions that would use lunar-derived propellants. L2 has also been proposed as a staging point for missions that might use a largely reusable chemical stage or electric propulsion. The high thrust delta V from the Space Station orbit to L2 (approx. 3.5 km/sec) is less than the delta V to GEO. It is not much less than the conjunction class trans-Mars injection delta V from LEO however. L2 staging will probably require substantial infrastucture in high orbits and may therefore be viewed as a longer term option that still requires study. Use of lunar-derived propellants (Ref. 4) will depend on the ratio of lunar to Earth launch costs and is still under study.

Delta V from LEO to low lunar orbit (4.13 km/sec) is almost the same as the LEO to GEO delta V (4.2 km/sec). As a first order approximation, we can therefore assume that a LEO based spacecraft that can retrieve a

Mars mission crew from GEO can also retrieve one from low lunar orbit or L2.

The high elliptical parking orbit requires the minimum insertion burn of a returning Mars spacecraft. The higher the apogee, the less the burn. Table 1 shows the insertion burns required for a number of orbits for conjunction and opposition missions. The best high thrust way to get to a high circular orbit is first to do an "Earth flyby" or insert into an ellipse with apogee at the desired circular altitude. Table 1 illustrates this, showing insertion delta Vs with and without flybys for a number of cases.

Figure 1 shows initial LEO mass versus round trip mass for a number of mission configurations. One extra ton carried round trip requires from 3.3 to 31.9 extra tons initially in LEO, depending on the mission trajectory and propulsion type. Recovery from a 24 hour ellipse without plane change,using LEO- based OTVs, costs roughly 2 metric tons for every ton recovered to 500 km circular LEO, depending on the scheme. It therefore pays in terms of initial mass in LEO to carry as little propellant and stage as possible for the Earth orbit insertion burn. To reduce overall mass in LEO, the parking orbit with the minimum insertion delta V requirement should be used. This means using as high an apogee as possible. How high this can actually be requires more study. The stability of the longer-period ellipses has been questioned. The maximum may be somewhere around a 48 hour period ellipse with perigee at 500 km.

The radiation belts may cause problems for high elliptical parking orbits. Only a limited number of passes through the belts can be tolerated by a crew at the end of a long mission during which high level radiation exposure may have already occurred. If the "storm shelter," needed during interplanetary flight for protection from solar flares, is placed in the ellipse, it may protect the crew during passage through the belts. This requires more study.

Figure 2 plots initial mass in LEO versus elliptical orbit apogee and period for a number of configurations. The knee in the curve is around the 12 hour period orbit for chemical propulsion. The nuclear propulsion (NERVA) cases are relatively flat for the entire range. All the curves are flat beyond 12 hour periods. The 24 hour period ellipse,

Fig. 1

Inital Leo Mass versus round trip Payload Mass

All cases-3 stages (TEI and EOI combined) except for aerobraking case.
Hybrid-first two stages CRYO (O2/H2), last Propane/O2, 24 hour, 500KM perapsis parking orbit at Earth and Mars.
All cases carry two landers (124mt) and one Mars OTV (31mt).

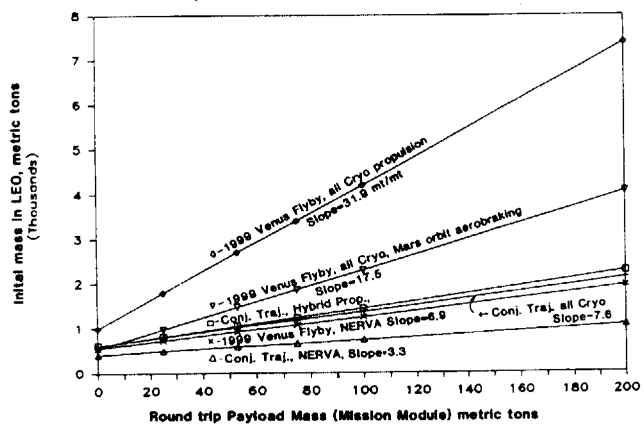


Fig. 2

Initial Mass in LEO versus return orbit

All cases-3 stages (TEI and EOI combined) except for aerobraking case, return 100% of 53mt mission module, carry two landers (124mt) and one MOTV (31mt), use 24 hour 500KM periapsis Mars parking orbit. Hybrid-TMI and MOI burns are all Cryo. (O2/H2), last two burns are O2/propane.

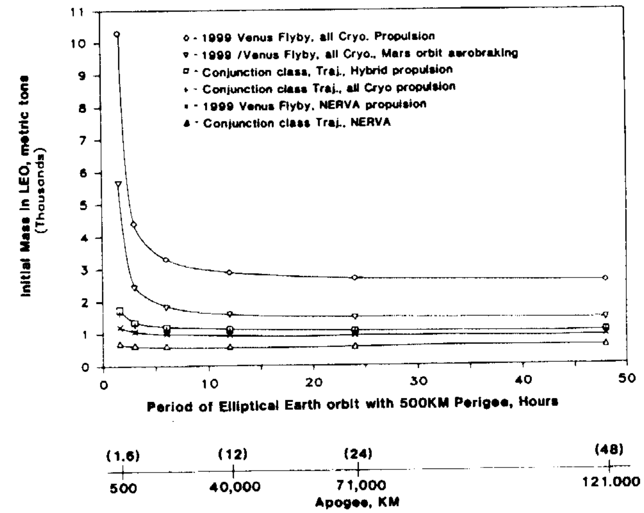
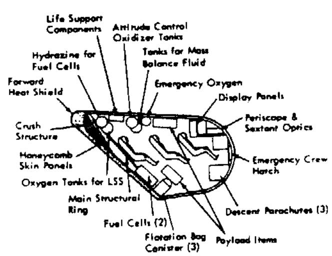


Fig. 3



Estimated Weights	
Crew and Seats	1,362 pounds
Controls	270
Guidance and Navigation	300
Communications	185
Science	912
Life Support	732
Electrical Power	659
Attitude Control	1,120
Recovery	870
Heat Shield	4,340
Structure	4,160
Growth & Contingency (15%)	2,240
Total	17,150 pounds (7,779 kg)

Biconic Earth Entry Module (from ref. 5)

SIX-MAN CREW
V_e = 60,000 fps

Fig. 4

Initial mass in LEO versus mass returned

All cases-3 stages (TEI and EOI combined) except for aerobraking case 24 hour, 500KM periapsis parking orbit at Earth and Mars.
All cases carry two landers (124 mt)and one Mars OTV (31 mt)
Hybrid-TMI and MOI burns are all Cryo. (O2/H2), last two burns are O2/propane.

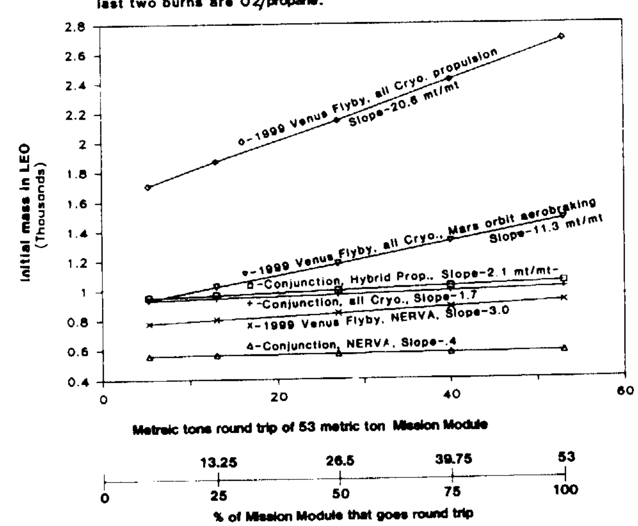


TABLE 1

DELTA V's FOR EARTH ORBIT INSERTION AND RETRIEVAL
1999 CONJUNCTION AND OPPOSITION TRAJECTORIES

Final Destination Orbit - 500 km (270 nm) circular, 28.5 deg., 1.58 hour period.

Insertion Orbit 500 Km perigee, 28.5 deg.			Delta V's 1	Delta V's 2	Delta V's 3
	Apogee Km	Period Hours	99 Opp. Insert. Delta V km/sec	99 Conj Insert. Delta V km/sec	Delta V from Ins. Orb. to Dest.km/sec
	121,000	48.00	3.55	0.91	2.87
	71,000	24.00	3.72	1.08	2.70
	40,000	12.00	3.99	1.35	2.43
	20,000	6.00	4.44	1.80	1.98
	7,870	3.00	5.2	2.56	1.22
	500	1.58	6.42	3.78	--
Direct Insertion into Circular Orbit					
	Altitude Km	Period Days			
(L2)	443,000	34.66	8.16	3.28	3.50
	121,000	5.23	7.57	2.95	4.07
(GEO Alt)	35,900	1.00	6.92	2.83 (28.5 deg. inclin.)	3.82
	20,370	0.5	6.67	2.91	3.37
(Space Station)	500	0.07	6.42	3.78	--
Insertion into Circular via Earth Flyby (and burn) at 500 km altitude					
(L2)	443,000	34.66	3.27	0.63	3.5
	121,000	5.23	4.75	2.11	4.07
(GEO Alt)	35,900	1.00	4.05	1.41 (28.5 deg. inclin.)	3.82

Note:
For 1999 Opposition, C3 inbound = 81; for 1999 Conj., C3 = 16 $(km/sec)^2$

with perigee at 500 km, is well beyond the knee in the curve, and has been used in a number of reference missions.

Direct entry into the Earth's atmosphere from the interplanetary trajectory requires no burn. Figure 3 shows a concept for a 7.8 metric ton direct entry capsule taken from reference 5. The large crew compartment flies on by Earth. The crew is only in the small capsule for a day or so. This approach results in the lowest initial mass in LEO of all and should not be discarded lightly. Its disadvantages include potential high g loads for a crew that may have just spent 2 to 3 years in zero g, no capability to quarantine the crew in the perhaps unlikely event Martian life is found and proves to be infectious on the long trip home, no capability for reuse of the large crew compartment or Mission Module, and the requirement to develop an additional entry vehicle.

Aerobraking into low Earth orbit avoids all but two of these problems. Initial studies indicate the g levels must still be high for a crew that has just experienced two to three years of zero g, and pre-entry burns are probably not a practical way to keep them down. If the crew habitat has significant artificial g, the g loads may not be a problem. The aerobrake, which may weigh 5 to 15 % of the aerobraked mass, must still be carried round trip, however, and will require significant additional development work. This aerobrake might also be used for Mars entry. The aerobraking option requires more study, and will be addressed in other papers.

Propulsive insertion into a high ellipse avoids all these problems at the cost of an Earth orbit insertion stage and the requirement to go after the crew and spacecraft with OTVs. It is therefore the leading contender at present.

HOW MUCH TO RECOVER

How much of the interplanetary spacecraft to recover? The options range from recovery to a refurbishment facility of an entire propulsion and crew module capable of single stage round trips, to direct entry into the Earth's atmosphere of a small crew module only as shown in Figure 3. Single stage options will probably require aerobraking at least at Mars and Mars orbit refueling, and are therefore longer-term options. The pros and cons of direct entry capsules are noted in the previous paragraphs.

Propulsive insertion of some fraction of the Mission Module and a trans-Earth/Earth orbit insertion stage into a 24 hour ellipse is considered in Figure 4, which shows the effect of inserting various masses for several reference missions. The increase in initial LEO mass/increase in inserted mass or slope of the lines in Figure 4 is not as great as the increase in LEO mass/total round trip mass (Figure 1). How much of the Mission Module is inserted into Earth orbit is not as important as how much the complete Mission Module and other round trip mass weighs. This other round trip mass could be propellant to lower the apogee of the ellipse. It must be carried round trip and inserted into the ellipse and is therefore very expensive, which makes it attractive to consider delivering it with an OTV to the returned spacecraft in high elliptical Earth orbit.

Since the actual Mission Module mass recovered is more a function of the economics of reuse than anything, it is beyond the scope of this work to define. This recovered mass will almost certainly be no more than 100 metric tons however, so a range from zero to 100 metric tons will be assumed.

METHODS OF RECOVERY FROM HIGH ELLIPTICAL EARTH ORBIT

Given the assumptions of a 24 hour period elliptical parking orbit and a mass range of zero to 100 metric tons, several methods for recovering this mass to the Space Station orbit can be proposed: 1) An unmanned OTV can dock with the spacecraft and propulsively return it to the Space Station orbit; 2) A manned or unmanned OTV can bring up an aerobrake to attach to the spacecraft, which then lowers apogee by aerobraking; 3) A manned or unmanned OTV brings up propellant to refuel the Earth orbit insertion stage and the spacecraft comes down propulsively; and 4) A manned OTV recovers the crew and mission artifacts and the spacecraft is left in orbit or deorbited to a controlled re-entry.

In the following analysis, a space-based aerobraked OTV, as shown in Figure 5, is assumed. This OTV has an empty weight of 7 metric tons, carries 42 metric tons of liquid hydrogen and oxygen that is burned at a specific impulse of 480 seconds, and carries an 8 metric ton crew module capable of carrying a crew of 8. It is assumed to be reusable and stackable as shown in Figure 6.

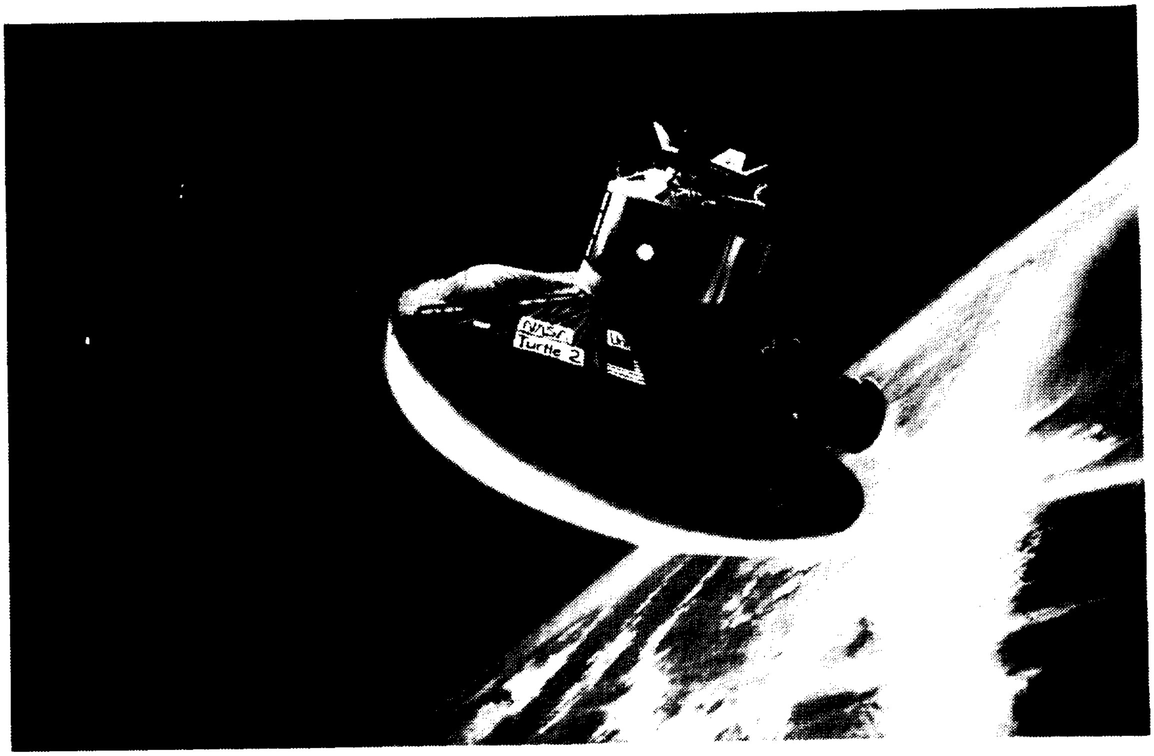


Fig. 5 Aerobraking OTV

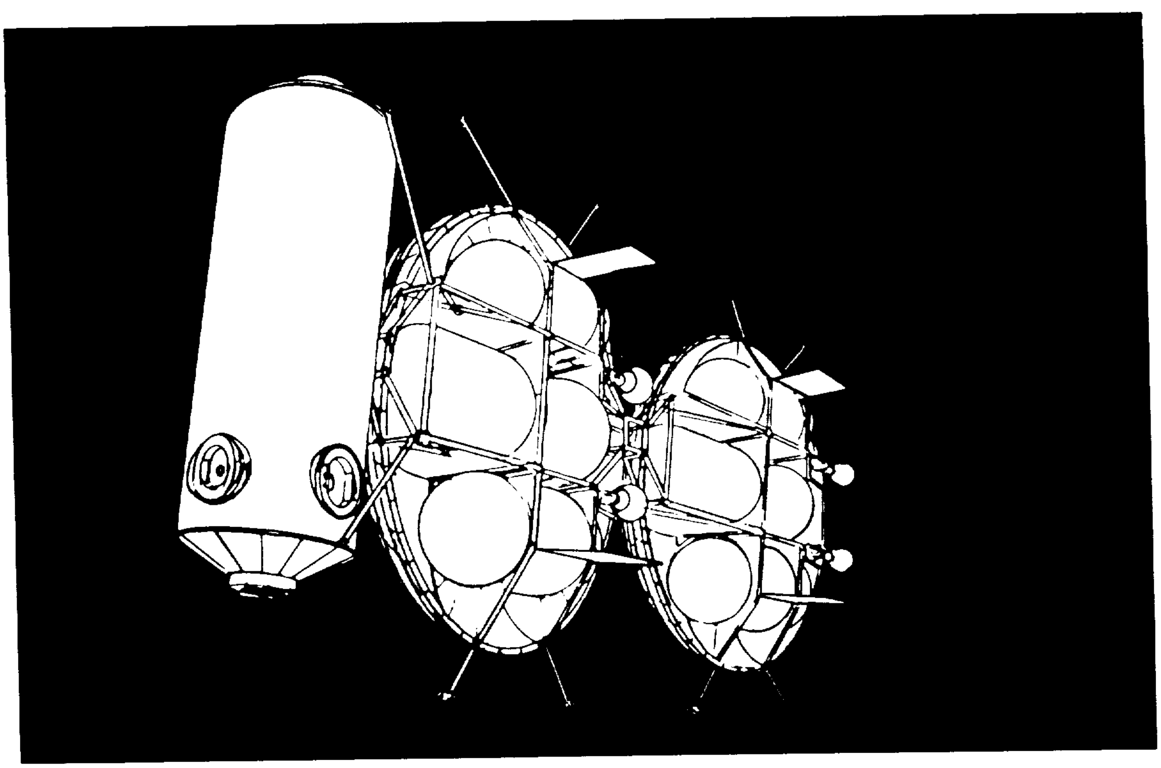


Fig. 6 Stacked OTVs

Table 2 shows a range of numbers for an unmanned OTV(s) docking with the Mars spacecraft and pushing it to LEO. One OTV uses 21 metric tons of fuel to deliver itself and a maximum of 20 additional metric tons of propellant in its own tanks from the Space Station orbit (500 km, 28.5 deg. circular) to the 24 hour ellipse (71,000 km x 500 km, 28.5 deg). One OTV can also deliver a second OTV with a maximum of 39 metric tons of propellant in its tanks to the 24 hour ellipse. The first stage OTV then aerobrakes back to LEO.

The last row in Table 2 shows the OTV propellant needed in LEO over the returned mass. For the heavier masses, this number is constant around 2.0. This means 2.0 metric tons of OTV propellant are needed in LEO for every 1.0 metric ton of Mars Mission Module brought back to LEO with the OTVs. Each metric ton of propellant placed in the 24-hour orbit can return approximately one metric ton of Mission Module to LEO from the 24 hour orbit. If this metric ton of propellant had to go round-trip to Mars it would have cost between 3.3 and 31.9 metric tons in LEO. By using the OTV-delivered propellant we are thus saving between 3.3-2 = 1.3 and 31.9-2 = 29.9 metric tons in LEO per metric ton of Mission Module recovered to LEO with this technique. This can be a good mass trade, particularly for the opposition class missions. The OTV sorties are not free however. A cost analysis is required.

The case in which a manned or unmanned OTV brings up an aerobrake to attach to the spacecraft has an even better mass trade, but introduces additional operational complexities and costs. One OTV can deliver an 8 metric ton (8 person) crew module, a 15 metric ton aerobrake (capable of aerobraking an entire 100 metric ton spacecraft), 7 metric tons of oxygen and hydrogen propellant for the Mars spacecraft or Mission Module to do perigee lower/raise manuevers, and an additional tank of 12 metric tons of propellant to bring itself and the crew module back propulsively to keep the returning Mars crew from experiencing high acceleration loads.

One OTV can handle the worst case aerobrake situation. The Mars spacecraft must be compact enough to be aerobraked however, and the aerobrake must be assembled in LEO. The total payload mass of the OTV is 42 metric tons. To deliver this the OTV uses 39 metric tons of fuel. For 100 metric tons recovered, the OTV LEO mass over recovered mass is

TABLE 2

UNMANNED OTV DOCKS WITH SPACECRAFT AND PROPULSIVELY RETURNS IT TO STATION

No. OTVS Required	1 (single stage)	1 stack of 2 (two stage)	1 stack of 2 plus 1	3 stacks of 2
Inserted Mass (MT)	7	42	50	100
Prop. to Return (MT)	11	39	50	95
OTV Prop. in Leo (MT)	31	81	106	203
OTV Prop. over Ins. Mass	4.43	1.93	2.13	2.03

24 hour ellipse parking orbit (71,000 x 500 km, 28.5 deg.)
500 km circular, 28.5 deg. destination orbit

TABLE 3

PROPELLANT A 42 MT CAPACITY OTV CAN DELIVER TO THE 24 HOUR ELLIPSE. OTV AEROBRAKES BACK TO LEO

	MANNED	UNMANNED
Delivered Prop in OTV Tanks	16	20
All delivered Prop in 2 mt mass external tank (not part of OTV)	35	43

roughly .8, better than 2.0 in the previous case. The cost to develop the aerobrake may be significant, however.

Table 3 shows the propellant which a manned or unmanned OTV can deliver to the Mars spacecraft, such that it can return itself propulsively to a space station compatible orbit. An extra (external) tank will be required for most cases. Table 4 shows the propellant that must be delivered for both manned and unmanned OTVs and for cryogens and storables. The manned LEO OTV propellant divided by the recovered mass ranges around 1.8 to 2.0 for cryogens and around 2.8 for storables. In terms of mass gain in LEO it is similar to the case where the OTV pushes the Mars spacecraft. Propellant transfer and tankage requirements will probably make it cost more however.

A single manned OTV can easily recover the crew and artifacts only, bring them back propulsively, and send a 100 metric ton spacecraft in the 24 hour ellipse to a controlled re-entry with a 200 m/sec push. It requires a full 42 metric tons of propellant.

In summary, the baseline case of a 50 metric ton Mission Module can be entirely recovered in several ways. It can be done with one OTV flight that delivers an aerobrake to it and recovers the crew. One OTV could also recover the crew and deorbit the spacecraft. Two OTV flights can deliver enough propellant to the mission module to allow it to utilize its own propulsion system to return to LEO. Three OTV flights (one stack of two plus one) can push it to LEO.

TABLE 4

OTV DELIVERS PROPELLANT

INSERTED (returned) MASS (MT)	7	42	50	100
		CRYOGENS		
PROP. REQ. TO RETURN (MT) (480 ISP)	6	33	39	79
		UNMANNED		
NO. UNMAN. OTV FLIGHTS TO DELIVER	1	1*	1*	2*
TOT. OTV PROP. MASS REQ.	16	65	75	155
		MANNED		
NO. MAN. OTV FLIGHTS TO DELIVER	1	1*	2*	3*
TOT. OTV PROP. MASS REQ.	24	72	95	181
		STORABLES		
PROP. REQ. TO RETURN (MT) 340 ISP	9	53	64	127
		UNMANNED		
NO. UNMAN. OTV FLIGHTS TO DELIVER	1*	2*	2*	3*
TOT. OTV PROP. MASS REQ.	23	109	128	249
		MANNED		
NO. MAN. OTV FLIGHTS TO DELIVER	1*	2*	2*	4*
TOT. OTV PROP. MASS REQ.	29	122	141	281

*Delivered Propellant is in extra external tank.

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