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COMPARISON OF MISSION DESIGN OPTIONS FOR MANNED MARS MISSIONS

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ABSTRACT

A number of manned Mars mission types, propulsion systems, and operational techniques are compared. Conjunction and opposition class missions for cryogenic, hybrid (cryo/storable), and NERVA propulsion concepts are addressed. In addition, both Earth and Mars orbit aero-braking, direct entry of landers, hyperbolic rendezvous, and electric propulsion cases are examined. A common payload to Mars was used for all cases. The basic figure of merit used was weight in low Earth orbit (LEO) at mission initiation. This is roughly proportional to launch costs.

INTRODUCTION

There are many ways to design a manned Mars mission. The optimum design depends a great deal on the long and short term goals of the program. These are at present officially undefined, but range from beating the Russians to Mars with a one landing program to permanent colonization. A program to carry large quantities of material to Mars over a long period of time will tend to settle on designs with minimum initial mass in LEO (includes vehicles and propellants) since Earth launch costs will eventually overwhelm development costs. A short term, one or two mission program, perhaps schedule driven, could concentrate on minimum development costs rather than minimizing LEO mass. The best design depends on the program. In the absence of clear direction, mission designers will produce designs that tend to fulfill their own personal view of what a manned Mars program should be. Since the authors of this paper favor a long term program and would like to see propulsion technology advance, minimum LEO mass is emphasized. Others may have different, but not at all incorrect views.

SCENARIOS

The basic scenario advanced in this paper is a Mars mission carrying two aerobraking landers/ascent stages of 62 metric tons total mass each, one Mission Module (MM) of 53 metric tons, and one Orbital Transport

Vehicle (Mars-OTV) of 31 metric tons. The spacecraft leaves a 500 km circular low Earth orbit, the basic Space Station orbit, and transfers to Mars. At Mars it boosts into a 24 hr ellipse (500 x 33,000 km) at the proper inclination so that perigee precesses to be lined up correctly for departure to Earth at the proper time. Once in Mars orbit the two manned landers descend to the surface while the MM and propulsion stages remain in elliptical orbit. The Mars-OTV is used by the crew to rendezvous with and explore the two Martian moons. At the end of this surface exploration, the two ascent stages (one on each lander) launch to low Martian orbit where the Mars-OTV meets them and transfers crew and samples up to the MMM. The ascent stages and the MOTV are then discarded. The propulsion stage(s) then return the MM to a 24 hr Earth ellipse (500 x 72,000 km) where it is met by an OTV from the Space Station.

MISSION TYPES

The above scenario was examined for a generic conjunction mission and opposition type Venus swingby missions for the years 1999, 2001, and 2005, as defined in Reference 3. In addition, an electric propulsion case and two hyperbolic rendezvous cases were included.

The conjunction mission uses a near Hohmann transfer from Earth to Mars, a one and one-half year wait at Mars for proper planetary phasing, and a near Hohmann transfer back to Earth. This is the minimum-energy mission with a total mission time of approx. 1000 days and flight opportunities every two years. Delta-V requirements vary somewhat between mission opportunities, but remain constant enough so that a generic Delta-V budget can be constructed for planning purposes.

The opposition missions require transfer to Mars, a stay time of 30 to 60 days, then a transfer back. Because of the phasing, non-Hohmann, high-energy transfers must be used. It has been found that a Venus swingby, either outbound or inbound, can substantially reduce the total energy requirements. Such a swing-by exists for virtually every mission opportunity every two years, but the variation in the three-body relationships creates large Delta-V variations between missions. Thus, each opportunity must be addressed as an entirely separate mission. These missions typically take around 700 days.

The electric thruster case gives high ISP but very low thrust. For low thrust the system (unmanned) spirals out from LEO to some high orbit such as the L2 Lagrangian point. The crew is then transported to the spacecraft via a high thrust OTV flight from LEO. The manned Mars stack then spirals out to Mars and slowly spirals down to low Mars orbit. The landers are dispatched and when the phasing is suitable the process is reversed to return to Earth.

When the power supply is sufficiently large, this reduces to a conjunction type mission with spirals at both ends. The time at Mars including spiral down, orbit operations, and spiral back up becomes the year and a half Mars stay time of the conjunction missions. Electric thruster mission times vary from a minimum of 3 years upward depending on the power source. Practical manned missions will require one megawatt or more of electrical power.

The hyperbolic rendezvous concept requires a launch from Earth carrying the landers and a MM. When Mars is reached, the system does not deboost into Mars orbit; instead, the landers separate and perform hyperbolic aerobraking entry maneuvers to landing sites while the Mission Module flies by Mars and is discarded. A second spacecraft with a second Mission Module leaves Earth at nearly the same time as the first spacecraft, but on a year and a half period trajectory that passes Mars 30 days after the first vehicle. The ascent stages that were landed from the first vehicle launch as the new MM passes by and perform hyperbolic rendezvous maneuvers with it. The crew must then ride the MM for one and a half orbits until it reintersects Earth. Mission time is three years, almost all of it in transit.

A modified version of this, the hyperbolic exchange, assumes a continuing manned base on Mars. The original vehicle with MM and landers is launched into the one and one-half year orbit, passing Mars. As it passes Mars the landers separate and do a hyperbolic entry and landing while, simultaneously the crew that had landed on the previous mission two years before launches to a hyperbolic rendezvous with the MM for the orbit and one-half flight back to Earth. In effect, a crew exchange takes place. Total mission time for a crew with this scenario is at least 5 years. Delta-V's for the various missions are given in Table 1.

TABLE 1

MISSION DELTA-V'S M/SEC

Mission Type	TMI	MOI	TEI	EOI
Conjunction Generic	3808	1666	1490	967
Opp. 1999 In-bound Swingby	4489	2757	1628	3725
Opp. 2001 In-bound Swingby	3792	1798	3633	1252
Opp. 2005 Out-bound Swingby	4400	3543	1673	1198
Low Thrust	13300	2600	8300	0
Hyperbolic Rend. Launch	3799	0	0	0
Hyperbolic Rend. Pickup	3843	0	81	1474
Hyperbolic Rend. Exchange	3843	0	81	1474

PROPULSIVE SYSTEMS

Hybrid

The hybrid system was used as a baseline. It consists of cryogenic liquid oxygen-liquid hydrogen (LO2/LH2) stages for trans-Mars injection (TMI) and Mars orbit insertion (MOI) and a LO2/propane "space storable" stage for trans-Earth injection (TEI) and Earth orbit insertion (EOI). This eliminates the problem of storing liquid H2 in the high heat environment of Mars planetary orbit, where additional cooling equipment to reduce propellant boiloff would be required.

All-Cryogenic

This system uses LO2/LH2 for all stages. This assumes that insulation and refrigeration are developed to allow long term (2 to 3 year) H2 storage.

NERVA

This nuclear rocket system uses nuclear engines with hydrogen as a reaction mass. Three engines of 75,000 lb. thrust each were used. All three are used for TMI to get the thrust/weight up to around .1 in order to keep gravity losses from being excessive. After TMI, one engine and all the empty hydrogen tanks are discarded. Engines 2 and 3 are used together to perform MOI. Engine 2 and the tanks emptied during MOI are then discarded. Engine 3 then performs TEI and EOI. Again, long term hydrogen storage is required. This also assumes that the NERVA engines can be started, shut down, and restarted several times while still maintaining their 10 hour total thrusting lifetime.

Electric Propulsion

High power, low thrust, high Isp ion engines are used for this system. Isp's from 3,000 to 20,000 seconds were examined requiring power supply sizes from .2 to 6 megawatts. Though ion engines with nuclear electric power is a reasonably well known case, any thruster and power processing system with specific mass in the 10 kg/kw range and primary power supply with specific mass as shown in Table 2 will provide equivalent performance. The stage characteristics and other parameters used are shown in Table 2. The electric propulsion design used only a single stage. The delta Vs shown in Table 1 for Low Thrust assume a spiral out to L2 and a transfer to Mars vicinity summed together as TMI, a spiral in to Mars (MOI), and a spiral out from Mars and transfer to Earth-Moon

TABLE 2

PROPULSION STAGE CHARACTERISTICS

Stage Type	All-Hybrid	Mer. Cryo	Ces. Nerva	Ion	Ion
Stage # 1					
Isp	468	468	825	3,000	20,000
A	0	0	11.5	*	*
B	0.0811	0.0811	0.15	0.1	0.1
M.R. O2/Fuel	7	7	0	0	0
Stage # 2					
Isp	480	480	825	0	0
A	0	0	11.5	0	0
B	0.1765	0.1765	0.18	0	0
M.R. O2/Fuel	7	7	0	0	0
Stage # 3					
Isp	370	480	825	0	0
A	0	0	11.5	0	0
B	0.0638	0.1765	0.18	0	0
M.R. O2/Fuel	3.5	7	0	0	0

Stage inert weight = A + B x (Propellant wt.)
A = Mass of power and propulsion system
B = Structure and tankage factor (dimensionless)
All masses in metric tons

Note: For large chemical propulsion stages such as these, the weight of the engines and control systems can be included in the massless parameter B. This assumes 1the number and/or size of the engines increases with increases stage size so that a constant thrust to weight is maintained.

* For electric propulsion, A = power parameter + power processing & thruster parameter)x(electric power). The power processing and thruster mass parameter used for all cases was 10 kgm/kw. An overall conversion efficiency of .7 was also used for all cases. The power parameter as a function of total power is shown below:

Power, kw electric	200	600	1000	3000	6000
Power parameter kg/kw	40	30	15	10	10

L2 (TEI). The spent stage is left at L2, and the crew is transfered back to Earth with an OTV.

FLIGHT OPTIONS

The software built for this study allows us to stack any given mission (opposition, conjunction, etc.) with any propulsive system and payload configuration and combine these with any of a large number of flight case options. These include:

- O All propulsive four stage operations
- O All propulsive three stage operations
- O All propulsive two stage operations
- O All propulsive one stage operations
- O Aerobraking at Mars--two stage
- O Aerobraking at Earth--one, two, or three stage
- O Aerobraking at Mars and Earth--two stage

(Note: The above three aerobraking cases consider aerobrake weight as a % of braked cargo to be percentage is a variable parameter.)

- O Separation of landers before MOI with the landers performing hyperbolic aero entry--three stage

The cases using aerobraking at Mars can reflect aerobraking to different Mars apoapses by simply changing the TEI delta V to reflect the lower ellipse.

RESULTS

The bulk of the study concentrated on the generic conjunction and the three opposition opportunities with the three standard propulsion systems--hybrid, all-cryo, and NERVA. Figure 1 shows the mass required in LEO for each of these three propulsion systems applied to all four of the standard missions. These were all-propulsive cases, each carrying the same reference cargo set. This chart immediately yields the following results:

O All-cryo does not yield substantially better performance than the more conservative hybrid case.

O With chemical propulsion, the all propulsive opposition missions are significantly more expensive than the conjunction missions. Aerobraking reduces this disparity in cost.

O The NERVA system shows a clear performance advantage for Mars planetary missions. This advantage becomes more and more marked as the mission energy requirements go up. Consequently, the NERVA system could offer a reasonably practical option of flying some of the short stay opposition missions during the early phases of Mars exploration.

O Provided multi-megawatt power supplies are available, electric propulsion is competitive with NERVA and high thrust conjunction class missions, but not as flexible.

Figure 2 shows the impact of discarding part of the MM before the EOI burn. Again, the impact is greater on the high energy missions. This is not generally a major impact but the savings in launch costs (at approx. $1 million per metric ton) warrant examination of the reuse value of the MM parts.

Figure 3 shows the impact of aerobraking at Mars if the vehicle is aerobraked to the same 24 hr period ellipse as in the propulsive case. Various values of aerobrake mass as a percentage of mass to be carried are shown. Only the hybrid propulsion system was examined. The non-aerobraked references are shown as marks on the y-axis. These data show that the overall performance is relatively insensitive to the aerobrake mass in the range considered.

Aerobraking yields substantial gains; the greatest gains being shown for the outbound Venus swingby cases, where encounter (MOI) velocities at Mars are high. Aerobraking can bring some opposition missions down to a reasonable departure weight. (The problem encountered is high acceleration during braking and its effect on the crew).

Figure 4 shows the impact of aerobraking as the apoapsis of the post-aerobrake orbit is reduced. For this comparison, only the conjunction and the 2005 opposition missions with hybrid propulsion were examined. The aerobrake weight used is 15% of the mass carried. Targeting an aerobrake to a very high apoapsis ellipse is difficult because the target velocity is so near escape that even a relatively small aero-exit error could cause loss of the vehicle. The apoapsis may have to be targeted to as low as 2000 km (500 x 2000 km) to guarantee a safe capture.

Nearly all of the aerobraking advantage for the conjunction mission is lost if a low Mars orbit is used (because of the required delta V increase for TEI). However, the absolute change with apoapsis altitude

Figure 1
BASELINE MISSIONS

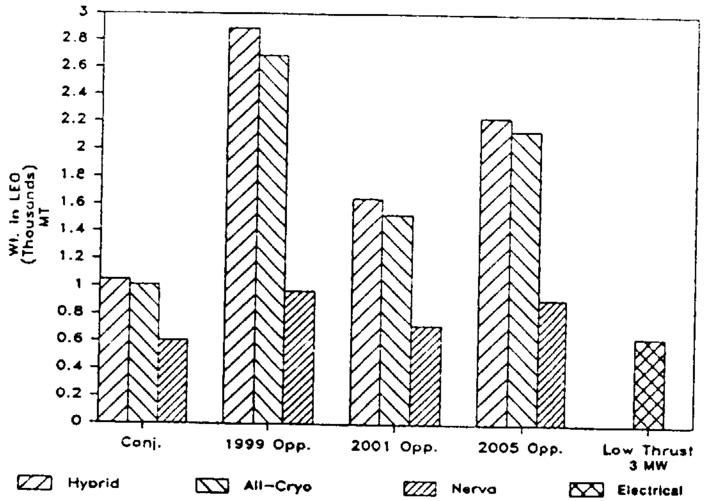


Figure 2
EFFECT OF DISGARDING MMM
50% OF MMM DISCARDED

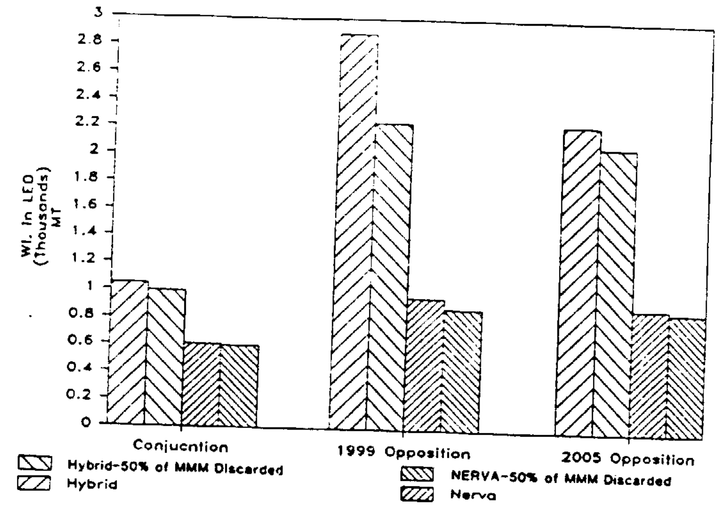


Figure 3
AEROBRAKE TO 24 HR MARS ORBIT
Hybrid Propulsion System (2 Stage)

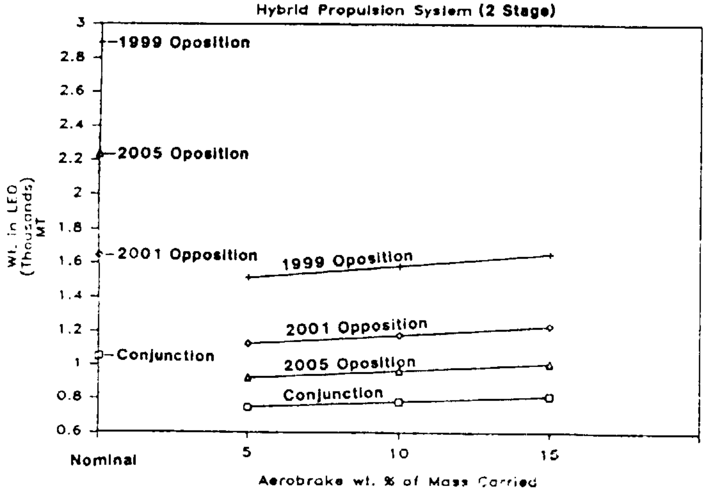
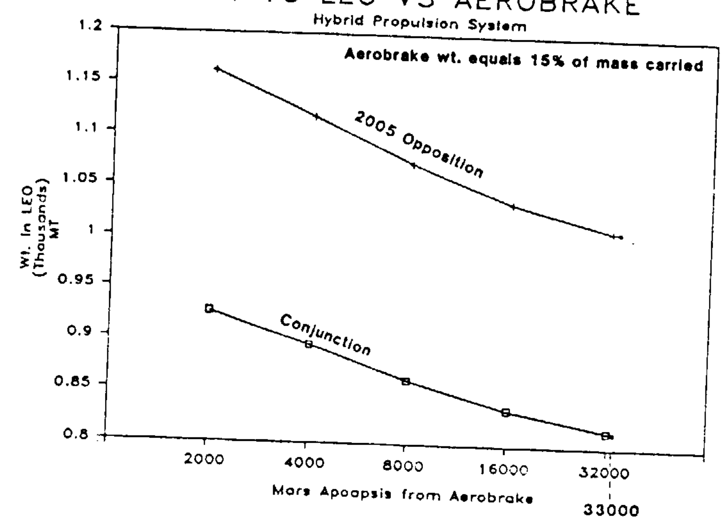


Figure 4
WT TO LEO VS AEROBRAKE
Hybrid Propulsion System



is nearly constant for both missions, so the 2005 opposition mission still shows a massive reduction from the all propulsive case.

Figure 5 shows aerobraking for different Mars apoapses, using a NERVA propulsion system. Again, the gains for the conjunction mission are minimal. The mass for the 2005 case is reduced by about a third; however, the potential advantage of aerobraking is not so great for the NERVA cases, which are already very efficient.

Figures 6, 7, and 8 show the sensitivity of the various missions to changes in lander weight (or cargo carried to Mars orbit and left). The three charts are for the three propulsion systems, hybrid, all-cryo, and NERVA.

Figures 9, 10, and 11 show the sensitivity of the missions to Mission Module mass (or mass carried round trip). The results of these figures for all 12 combinations are summarized in Table 3 as equations of the form: Initial weight in LEO = A ÷ B x (Lander & Mars-OTV Weight) ÷ C x (Mission Module Weight).

Figures 12 and 13 compare various aerobraking modes for the conjunction and the 2005 opposition cases with hybrid and NERVA propulsion. The most notable item is the relative effectiveness of releasing all landers pre-MOI and letting them aerobrake either to direct landing or to a low orbit to await landing site availability. Since the landers are designed for aero-entry already, it may prove relatively inexpensive to do this. Entry g levels may be high however.

Figure 14 shows the crew time, or the time the crew spends in the spacecraft from L2 departure to L2 return, versus power supply for the electric propulsion case. This defines the power requirement for each case since flight times should be kept below four years. Combined with Figure 15, which shows initial mass in LEO versus power, the two figures show that more than one megawatt of electric power will be needed. The lowest Isp cases have short trip times for low power, but Figure 14 shows their LEO masses are approaching the NERVA (600 metric ton) and conventional chemical conjunction (1,000 metric ton) cases. One 3,000 second case with a reduced payload of one lander and no MOTV might be performed with 600 kw. The low thrust cases must provide substantial LEO mass savings to offset the additional development costs; however, if large

Figure 5

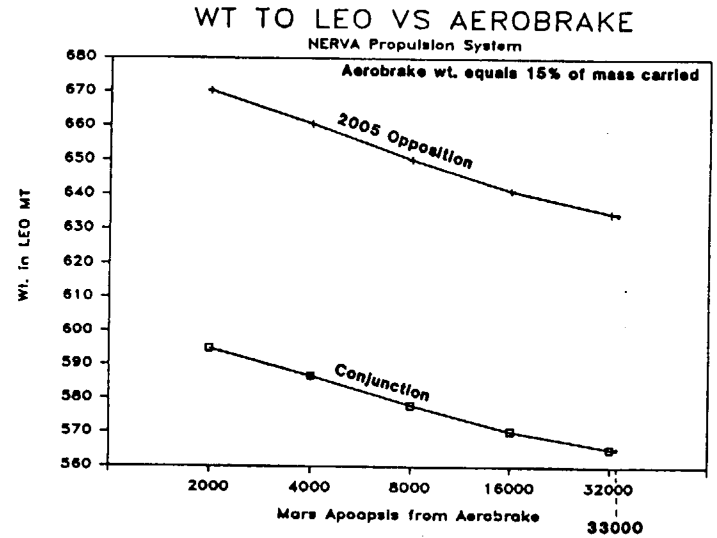


Figure 6

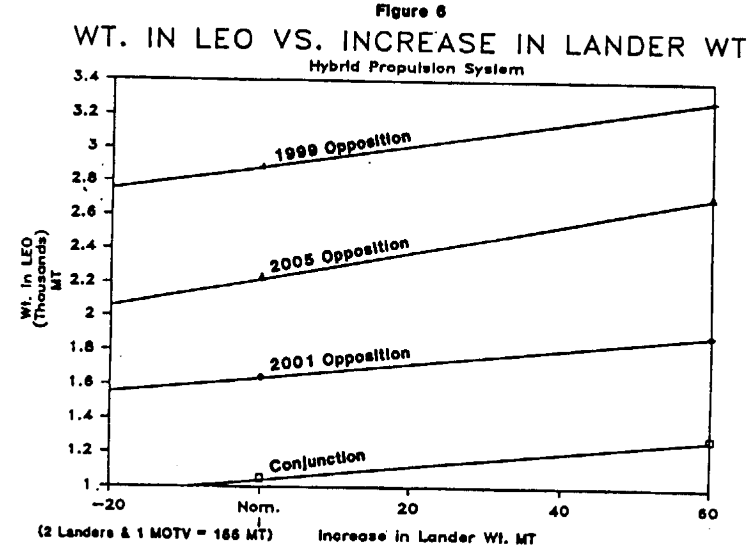


Figure 7

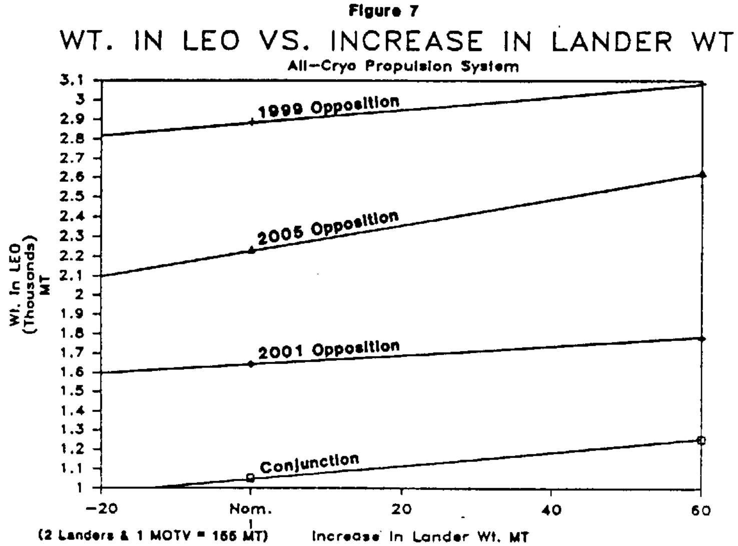
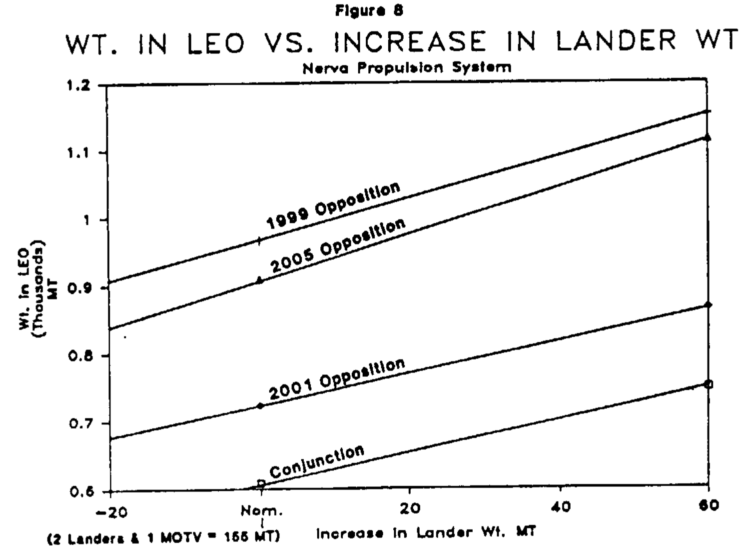


Figure 8



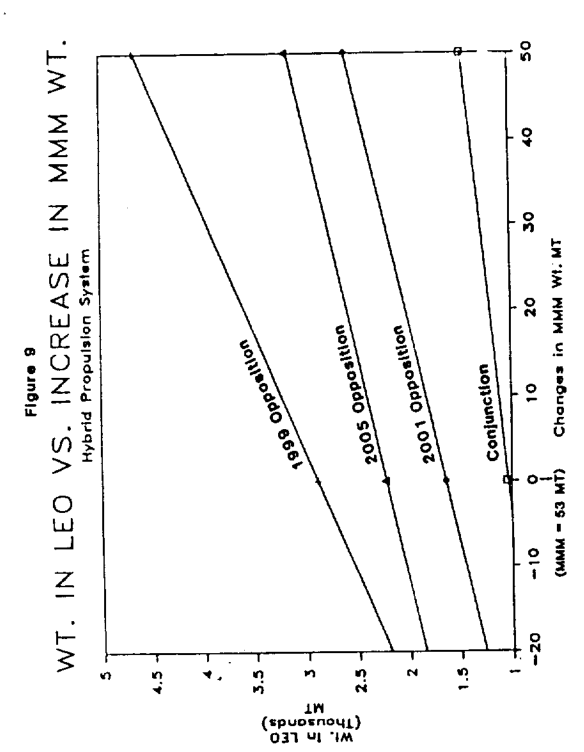


Figure 9
WT. IN LEO VS. INCREASE IN MMM WT.
Hybrid Propulsion System
1999 Opposition
2005 Opposition
2001 Opposition
Conjunction
Wt. in LEO (Thousands) MT
Changes in MMM Wt. MT
(MMM = 53 MT)

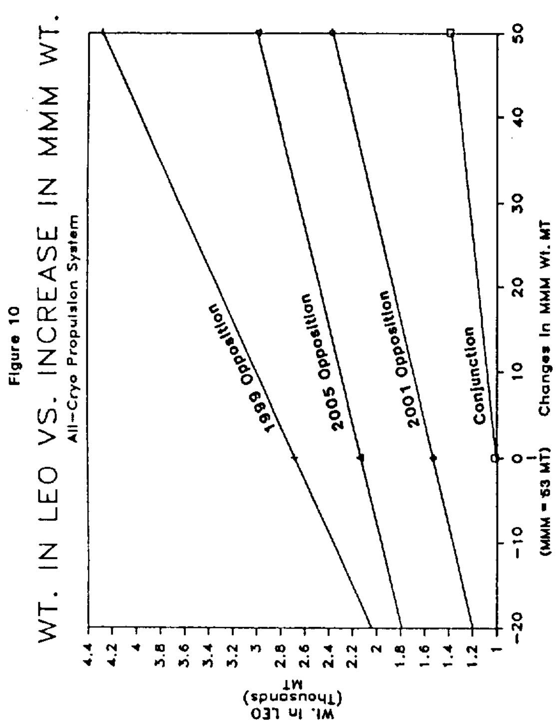


Figure 10
WT. IN LEO VS. INCREASE IN MMM WT.
All-Cryo Propulsion System
1999 Opposition
2005 Opposition
2001 Opposition
Conjunction
Wt. in LEO (Thousands) MT
Changes in MMM Wt. MT
(MMM = 53 MT)

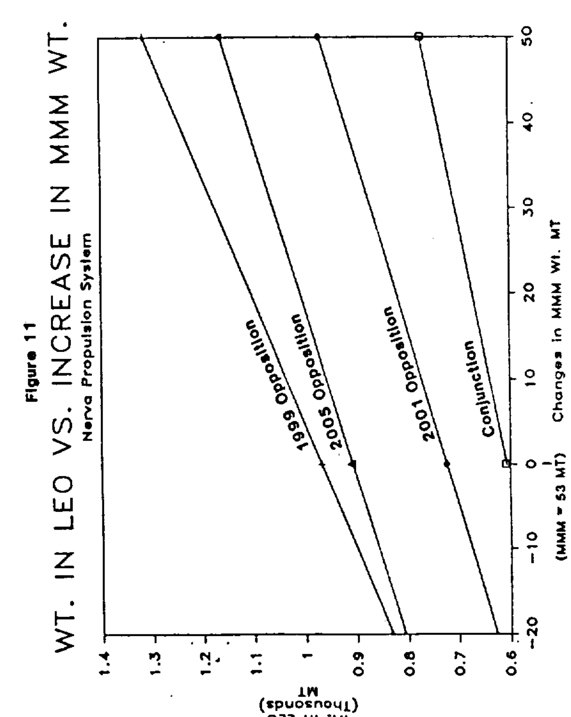


Figure 11
WT. IN LEO VS. INCREASE IN MMM WT.
Nerva Propulsion System
1999 Opposition
2005 Opposition
2001 Opposition
Conjunction
Wt. in LEO (Thousands) MT
Changes in MMM Wt. MT
(MMM = 53 MT)

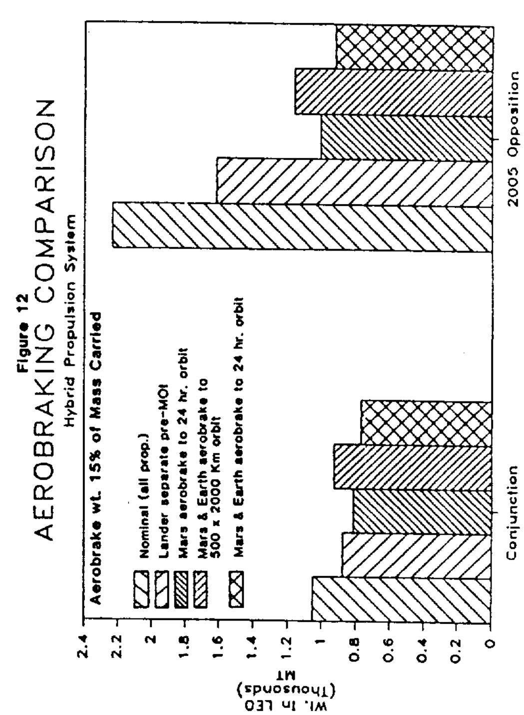


Figure 12
AEROBRAKING COMPARISON
Hybrid Propulsion System
Aerobrake wt. 15% of Mass Carried
Nominal (all prop.)
Lander separate pre-MOI
Mars aerobrake to 24 hr. orbit
Mars & Earth aerobrake to 500 x 2000 Km orbit
Mars & Earth aerobrake to 24 hr. orbit
Wt. in LEO (Thousands) MT
Conjunction
2005 Opposition

TABLE 3

WEIGHT IN LEO AS A FUNCTION OF PAYLOAD
TO MARS AND MM ALL RETURNED

Wt. in LEO = Empirical A ÷B x (lander & Mars-OTV) + (C x MM)

	Parameters	A	B	C
Conjunction Missions	Hybrid	A = 0	B = 3.94	C = 8.28
	Cryo	A = 0	B = 3.94	C = 7.56
	Nerva	A = 86	B = 2.25	C = 3.26
1999 Opposition	Hybrid	A = 0	B = 6.42	C = 35.73
	Cryo	A = 0	B = 6.42	C = 31.94
	Nerva	A = 140	B = 2.97	C = 6.93
2001 Opposition	Hybrid	A = 0	B = 4.07	C = 19.06
	Cyro	A = 0	B = 4.07	C = 16.92
	Nerva	A = 105	B = 2.30	C = 4.93
2005 Opposition	Hybrid	A = 0	B = 7.93	C = 18.96
	Cyro	A = 0	B = 7.93	C = 17.14
	Nerva	A = 100	B = 3.32	C = 5.12

A = Parameter relating required LEO Weight to NERVA systems Wt.

B = Parameter relating required LEO weight for systems carried one way.

C = Parameter relating required LEO weight for systems carried on round trip to Mars.

Figure 13
AEROBRAKING COMPARISON
Nerva Propulsion System

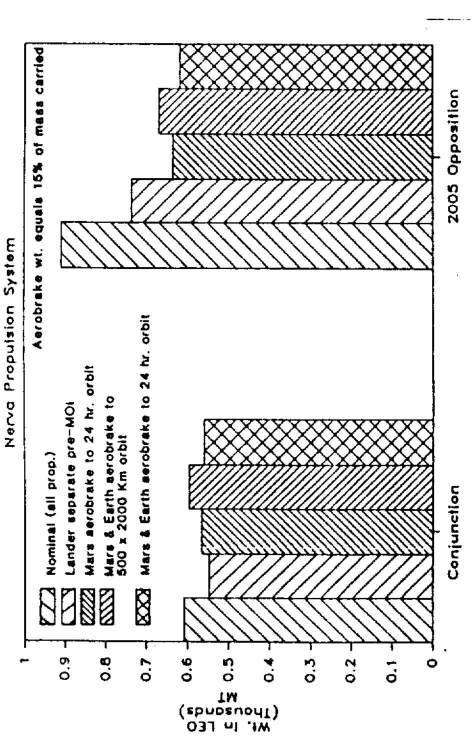


Figure 14
APPROXIMATE CREW TIME VS. POWER
53 MT Mission Module Carried Round Trip

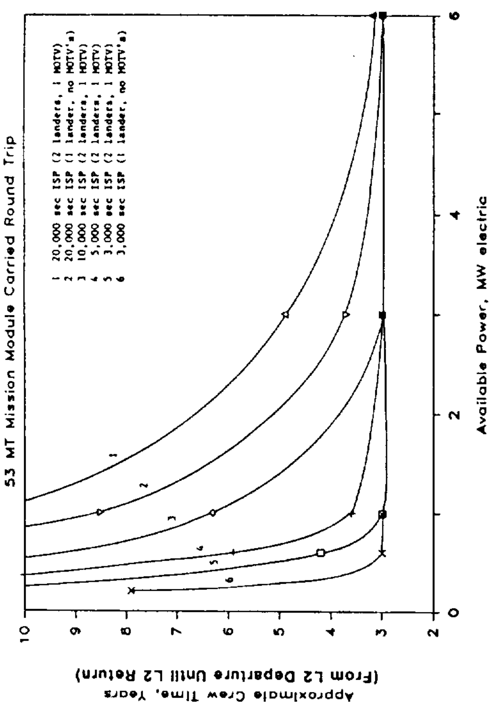


Figure 15
INITIAL LEO MASS VS. POWER SUPPLY
53 MT Mission Module Carried Round Trip

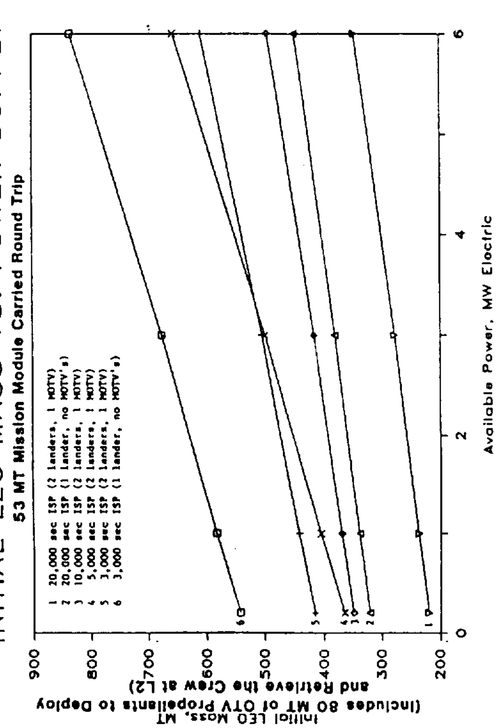
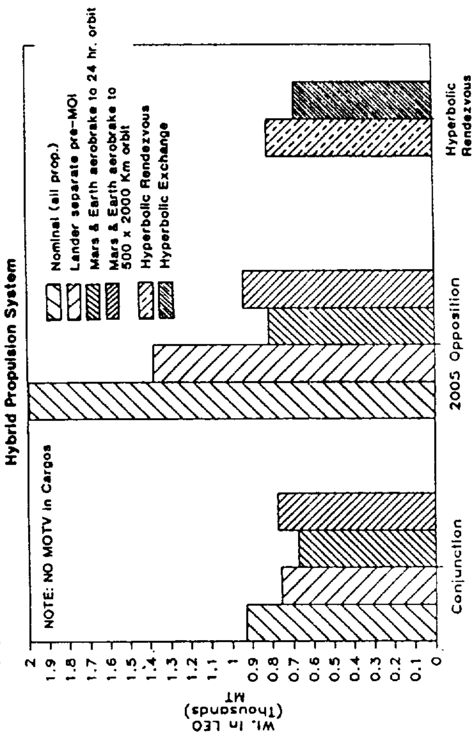


Figure 16
COMPARISON WITH HYPERBOLIC RENDEZVOUS
Hybrid Propulsion System



power supplies are developed separately, the low thrust opportunities will be highly competitive.

Figure 16 compares several aerobraking cases with the hyperoblic rendezvous schemes for hybrid propulsion. For this figure the Mars-OTV was removed from all cases to make a one-to-one comparison possible and the hyperbolic rendezvous landers were increased from 62 metric tons each to 90 metric tons (Ref. 1) each to account for the extra propellant required in the ascent stages to reach the hyperbolic outbound velocities. The hyperbolic case requires less mass than the opposition mission, but the comparison should be made with the conjunction missions since the total mission times are nearly the same (3 years). For hyperbolic rendezvous, nearly all the time is in interplanetary transfer, while for the conjunction missions, half of the time is at Mars. Hyperbolic rendezvous shows some weight advantage; however, nearly the same gain can be achieved in the conjunction case by simply staging the landers pre-MOI and doing a hyperbolic entry. This is much simpler than the hyperbolic landing and ascent required of the other case. Significant risk may be associated with the hyperbolic ascent and rendezvous.

GENERAL CONCLUSIONS

Advanced technology propulsion should be pursued vigorously to support a long term Mars program. Given the assumptions used in this paper, NERVA appears to yield an advantage even in the minimum energy cases and may provide the flexibility of flying the higher energy mission options. This advantage may become more pronounced as high energy missions to destinations past Mars are contemplated. This conclusion was also reached by workers of the late 60s (Ref. 1). Reference 1 documents the last large, overall systems level study done on a manned Mars mission/program on NASA contract.

The NERVA program, canceled in 1970, was designed with a manned Mars mission in mind. However, there were several problems which are assumed solveable in this paper.

o The old NERVA specific impulse estimate of 900 seconds was degraded to the 750 second region by erosion problems of the graphite core elements and by the propellant losses needed to cool the reactor after each burn. This paper assumes an Isp of 825 seconds.

O The inert shielding mass was high. This paper assumes a shield and reactor mass of 11.5 metric tons per stage. Changes in this can significantly alter the results. Formidable operations problems for manned operations in the vicinity of NERVA also would exist.

O The low density of the hydrogen propellant (4.4 lbm/ft^3) compared to O_2/H_2 (22-25 lbm/ft^3) resulted in higher cost per unit mass for delivery.

O No mission model large enough to absorb the development costs and still make the old NERVA program pay existed.

O Environmental and political/emotional impact of testing were severe.

O A "nuclear safe altitude" is not well defined. This paper assumed the NERVA could depart from a 500 km circular orbit. If this changes radically, the results may also change.

Aerobraking is worth continued investigation, particularly if no advanced space propulsion is available.

Conjunction class missions can be flown for reasonable weights even with chemical all-propulsive cases. However, either the NERVA or aerobraking is necessary to make the opposition missions a practical alternative.

Electric propulsion also offers weights in the NERVA range, but with less flexibility. Its feasibility hinges on the practicality and cost of megawatt level electric power supplies, which need to be determined.

REFERENCES

1. Stump, William R., Babb, Gus R., and Davis, Hubert P., Mars Lander Survey, Eagle Engineering Inc., Houston, Texas, NASA JSC Contract # NAS9-17317, presented at the Marshall Space Flight Center Mars Workshop, June 10-14, 1985, Huntsville, Alabama.

2. Integrated Manned Interplanetary Definition Study (IMIS), 1968, Boeing, Seattle, Contract No. NAS1-6674, 6 volumes.

3. Young, A. C., Mars Mission Concepts and Opportunities: In Manned Mars Missions, NASA Report M0002.