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MARS LANDER SURVEY

William R. Stump
Gus R. Babb
Hubert P. Davis

Eagle Engineering
Houston, TX

ABSTRACT

The requirements, issues, and design options are reviewed for manned Mars landers. Issues such as high l/d versus low l/d shape, parking orbit, and use of a small Mars orbit transfer vehicle to move the lander from orbit to orbit are addressed. Plots of lander mass as a function of Isp, destination orbit, and cargo up and down, plots of initial stack mass in low Earth orbit as a function of lander mass and parking orbit, detailed weight statements, and delta V tables for a variety of options are included. Lander options include a range from minimum landers up to a single stage reusable design. Mission options include conjunction and Venus flyby trajectories using all-cryogenic, hybrid, NERVA, and Mars orbit aerobraking propulsion concepts.

REQUIREMENTS

A manned Mars lander or Mars Excursion Module (MEM) will be one of, if not the major cost item in a manned Mars mission program. The nature of the program will determine the requirements for the lander. The major questions are: 1) How many landings or missions are to be flown, or what is the overall scope of the program? 2) How long must the lander support a crew on the surface? and 3) Must major cargo items be landed?

A short program with only two or three Apollo style landings would be required to support a crew for only a few weeks or a month on the surface, and land only a small amount of cargo. Cost would probably be the major driver. Only approximate guidance and navigation might be adequate.

A 20 mission program might require a lander that could spot-land, grow to support a crew for 100s of days on the surface, take advantage of surface propellant production, and perhaps land significant cargos, such as a surface base. Performance, which would be important in long term costs, might well be the driver.

The program is not defined at present, so we must look at all the options. The lander will be expensive and we only want to design one, and may only get the chance to design one, so the program must be carefully defined at the start.

It may be possible to design a Mars lander that can also be used on the Moon[1].

	Lunar	Mars
Descent Delta V, km/sec	2.08	1.23
Ascent Delta V, km/sec	1.91	4.84 minimum
		6.00 typical

Since the Mars lander ascent tanks will not be full when landing on the Moon, the descent tanks, sized for a Mars landing, may be able to handle lunar descent. Reference 1 proposed a lunar surface landing as part of a MEM test program.

ISSUES

The lift/drag shape of the lander is a major issue. Two basic families of shapes have been proposed, the low lift/drag (l/d) ratio or Apollo Command Module shape, and the high l/d or lifting body shape. Figures 1 through 4 show proposed low l/d shapes. Figures 5, 6, 7, and 8 show different high l/d shapes.

The low l/d shape is roughly 10 % lighter (Ref. 1) than typical high l/d designs. The low l/d lander is easier to build and test and therefore less expensive, and can accommodate growth more easily. The low l/d shape may be more easily built to land on the Moon. The low l/d shape may not be capable of direct entry into the Mars atmosphere from a trans-Mars trajectory (if this is a desired requirement), and may be more difficult to spot-land. Landing accuracy problems may be overcome to some extent by additional hover propellant.

Figure 9 shows a concept for a Mars base in a water-eroded canyon that would require spot-landing capability. Such a difficult landing site may be a desired target, because of the possibility of fossils or other evidence of life in those locations.

The high l/d shapes have a wider entry corridor, a much bigger footprint, and may be easier to spot-land. There is a problem keeping the g forces on the crew "eyeballs in" during both entry and ascent, however, without drastic measures. The high l/d shapes can enter directly from the interplanetary trajectory to the surface.

Fig. 1 Rockwell low l/d MEM
l/d-.5

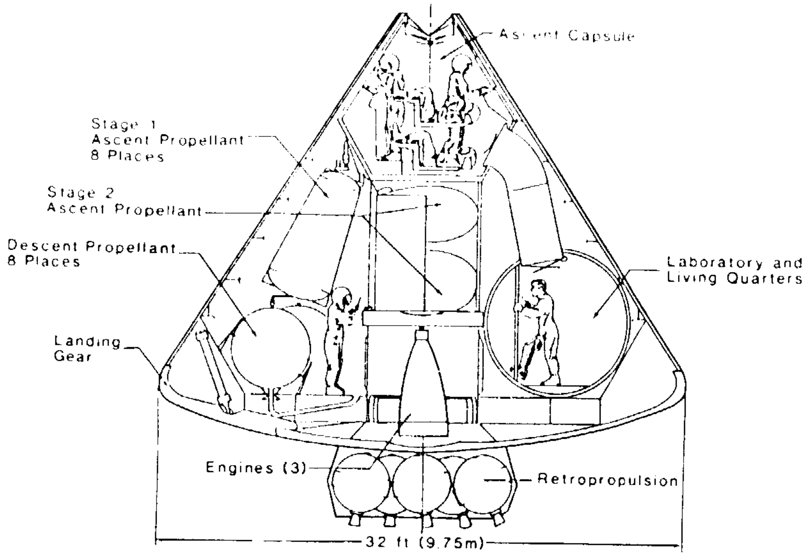

Fig. 2 Rockwell low l/d MEM, Mission Phase Configurations (from Ref.1)

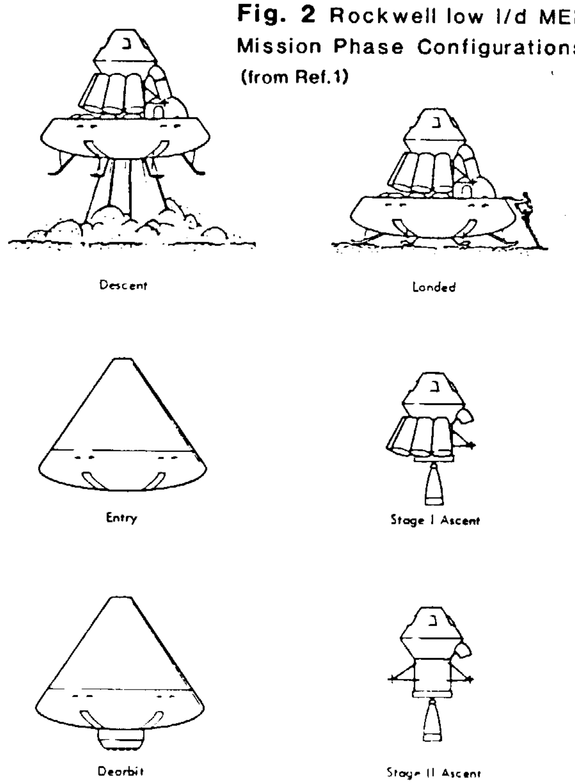

Fig. 3 Rockwell low l/d MEM with MSFC updates, Three View

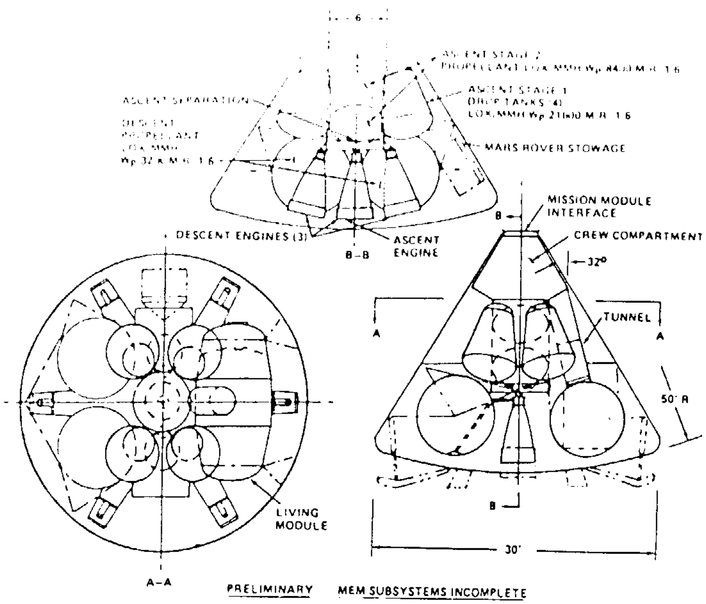

Fig. 4 MSFC low l/d concept with open afterbody

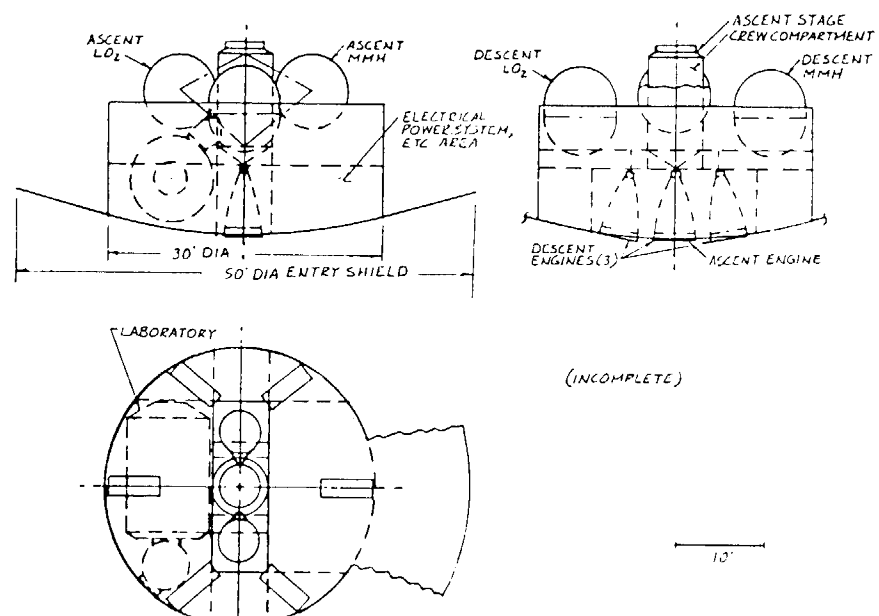

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Fig. 5 Rockwell lifting body MEM l/d-1.0, wings drop off before landing. (from Ref.1)

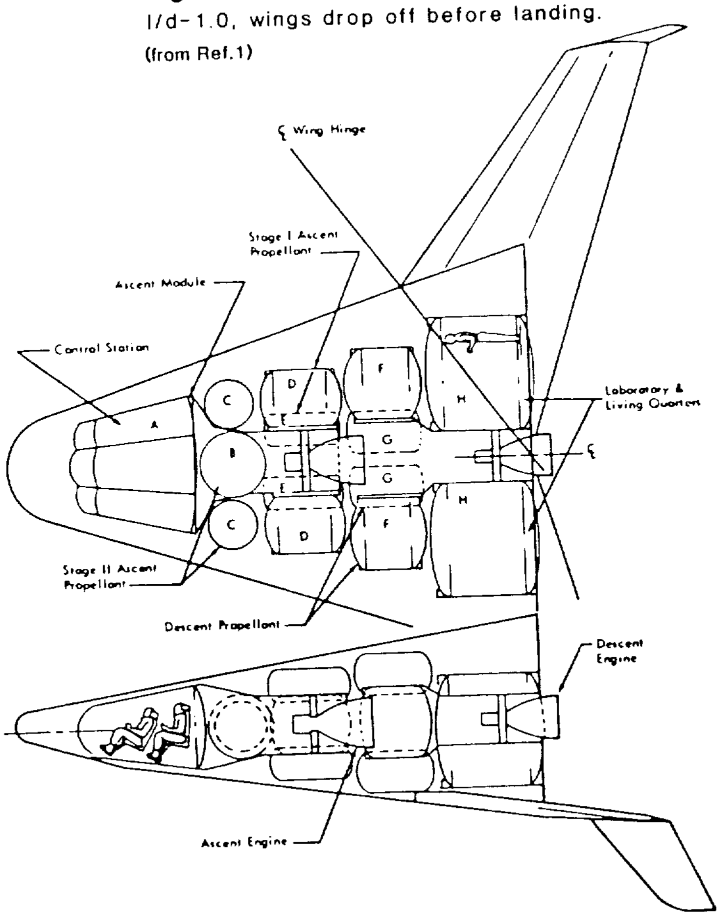

Fig. 6 Rockwell lifting body MEM ascent (from Ref.1)

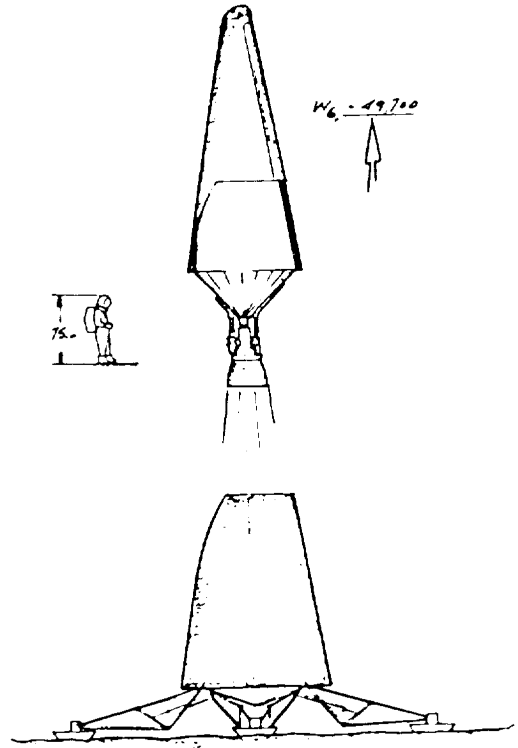

Fig. 7 Case for Mars II Bent Biconic Concept-uses surface produced propellants. (from Ref.4)

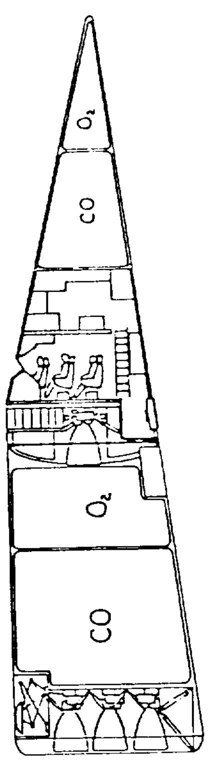

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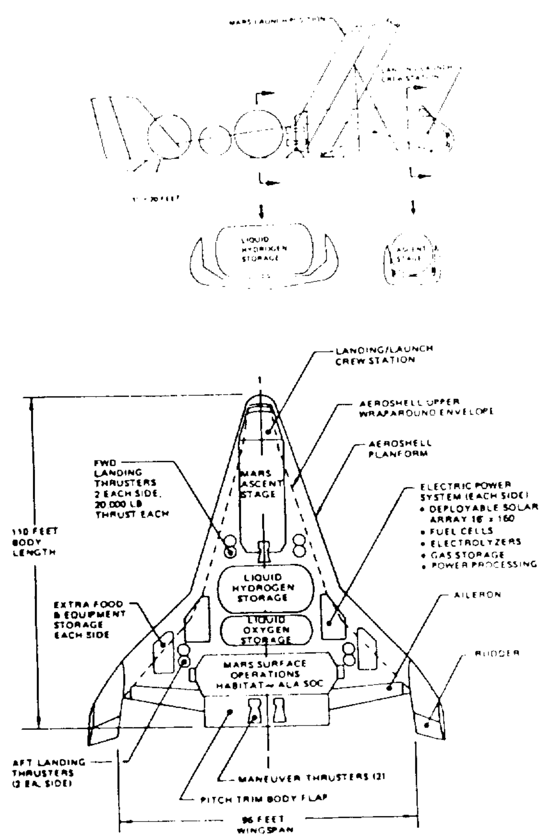
Fig. 8 Open Afterbody high l/d MEM (from Ref. 2)



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The most comprehensive study of manned Mars landers to date (Ref. 1, 1967), which did comparison designs of both high and low l/d shapes (Figures 1 and 5), chose the low l/d as a baseline. This was based on cost, testing requirements, and simplicity, and the absence of mission requirements that might dictate another choice (such as a requirement for direct entry). Since the body of data Rockwell subsequently generated (Ref. 1) on a low l/d design is extensive, and the mission requirements have not been defined much better since 1967, this paper uses the low l/d shape as a baseline for calculation purposes. To get high l/d numbers, add roughly 10% to the gross weights in the graphs and tables.

Another issue of significance is Mars parking orbit: low circular (500 km), high elliptical (24 hour), or none (direct entry from the interplanetary trajectory for the lander, and hyperbolic rendezvous with a passing interplanetary spacecraft at departure). The lander is insensitive to entry parking orbit (given a low perigee or a low circular orbit; this is not true for high circular orbit), in terms of mass, since it uses essentially an aerobraked entry. G levels for direct entry and entry from the elliptical parking orbits may be high, however. Ref. 1 predicts g levels of 4.5 for high elliptical versus 2 for low circular entry. This may make a significant difference for a crew that has been in zero g for six months or more.

The higher the orbit the lander must ascend to, the greater its initial mass. Figure 10 plots lander entry mass versus destination orbit for a variety of possible landers. The difference between low circular and hyperbolic escape values is only a factor of two or so. Figure 11 shows the effect of high elliptical and low circular parking orbit on initial mass in LEO for a variety of propulsion and trajectory schemes. The high elliptical parking orbit reduces Mars orbit insertion and trans-Earth insertion burns by over a km/sec each. This vastly overwhelms the effect of lander mass changes and can lead to a reduction in initial mass in LEO by factors of 1.3 to 2.0, depending on the mission propulsion and trajectory. So, based on LEO mass, the high elliptical parking orbit is better than a low circular orbit.

A small Orbital Transfer Vehicle (OTV) can also be used to ferry the MEM ascent stage from low circular Mars orbit to high elliptical Mars orbit. This small stage could result in savings of 10 to 20% of initial

Figure 9

Mars Base in a Canyon, spot landings required

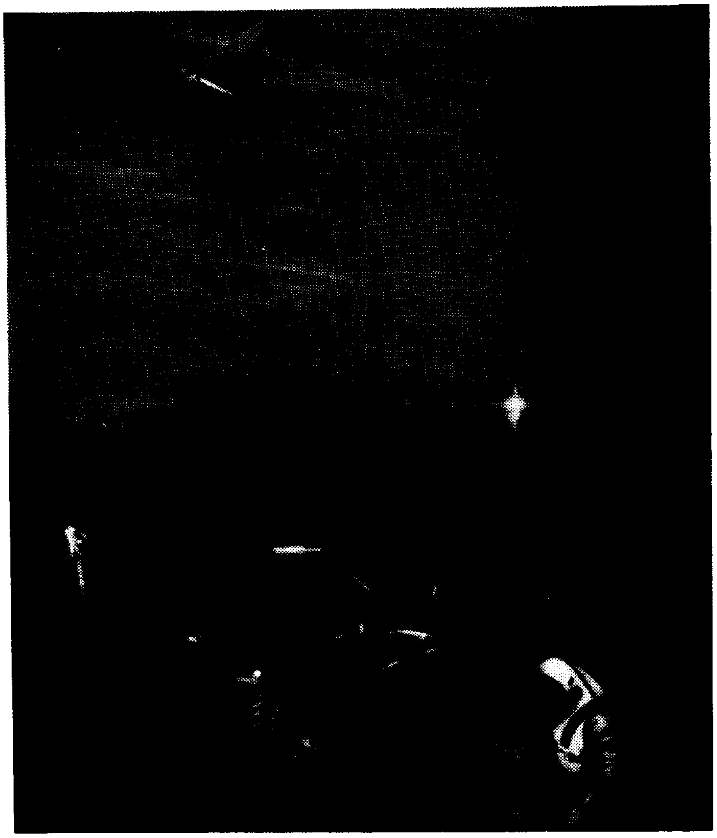
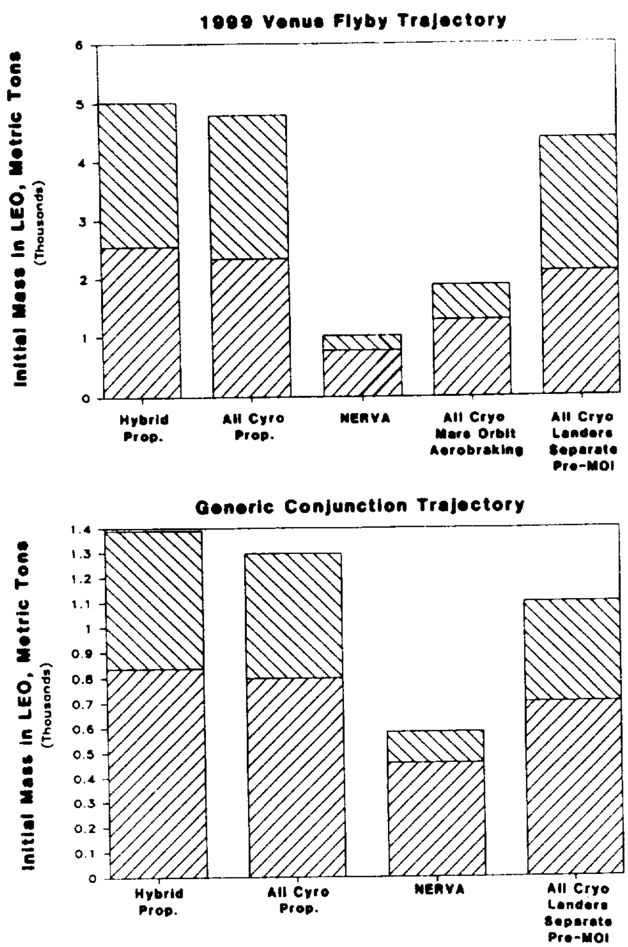


Figure 11

Initial Mass in LEO for 500KM circular and 500KM X 32,963KM (24 hour) Mars parking orbits.

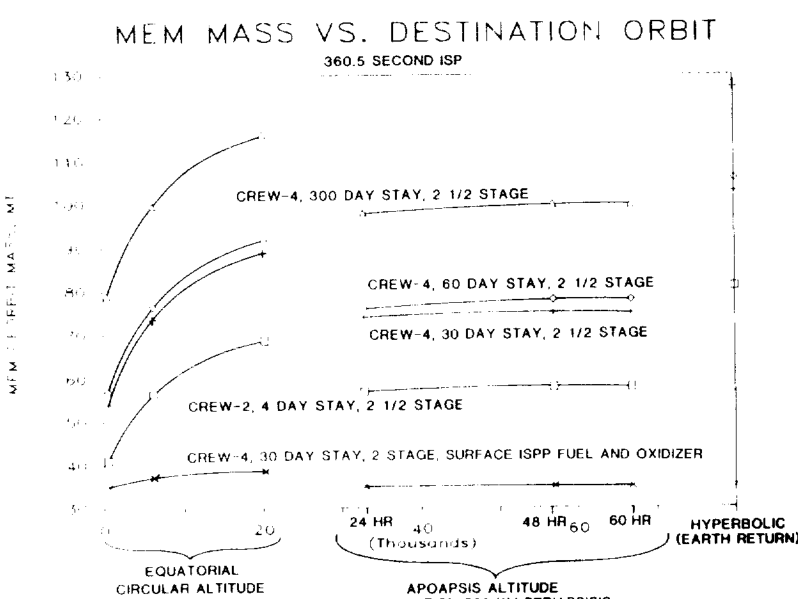


-500KM circular Mars parking orbit 55 metric ton lander (one)

-24 hour Ellipse parking orbit 70 metric ton lander (one)

All cases use a 53 metric ton Mission Module, 360.5 sec ISP landers and all carry a 31 metric ton MOTV. All cases are 3 stage, last stage does TEI and EOI. Hybrid-first two stages are all cryo. (H2/O2), last stage-O2/ Propane

Figure 10



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MEM + OTV mass in high elliptical Mars orbit compared to a one and one half stage MEM capable of ascending directly from the surface to high elliptical orbit. The cost of the OTV would probably overshadow the mass savings however, unless the OTV was required for another purpose, such as to visit Phobos and Deimos.

The Ref. 1 design uses no chutes or ballutes. That report concludes that this reduces the development cost substantially, but makes the lander 5 to 10% heavier. Figure 12 plots initial stack mass in LEO as a function of one-way payload mass to Mars (MEM + OTV mass) for a variety of cases. Note the slopes. One extra metric ton of lander and/or OTV mass costs 2.3 to 6.4 metric tons in LEO, depending on the propulsion and trajectory scheme.

Figure 13 plots lander mass versus specific impulse for a variety of cases. The cargo lander is insensitive to specific impulse, indicating a one way lander using solids might be possible. The MEM using surface-produced-propellant is also insensitive, indicating the proposed CO/O_2 propellant, whose Isp may be less than 300 seconds is feasible. The CO/O_2 propellant may be easy to produce from the carbon dioxide atmosphere of Mars.

Figure 14 plots MEM deorbit mass versus cargo mass down. The problem of a cargo lander will be packaging in an aeroshell. Figure 15 shows a lunar cargo lander unloading an 18 metric ton Space Station Common Module, postulated to be the largest and heaviest cargo to be landed on the Moon (Ref. 3). Figures 4 and 8 (from ref. 3) show low and high l/d concepts with open afterbodies that could accomodate such a cargo.

Figure 16 shows MEM deorbit mass versus ascent cargo mass for several cases. To lift tens of tons off the surface will strongly drive the design towards surface propellant production. Table 1 shows the delta Vs used to produce the plots discussed below.

CONFIGURATIONS

Figure 3 shows the 1967 Rockwell low l/d design with recent updates provided by the Marshall Space Flight Center (MSFC) group, which includes a different engine design and propellant. The weight statement provided in reference 1 with MSFC updates was extrapolated with scaling equations and other software to produce Tables 2 and 3 and Figures 11 through 16.

Initial Mass in LEO versus one-way payload to Mars
24 hour parking orbit at Mars and Earth

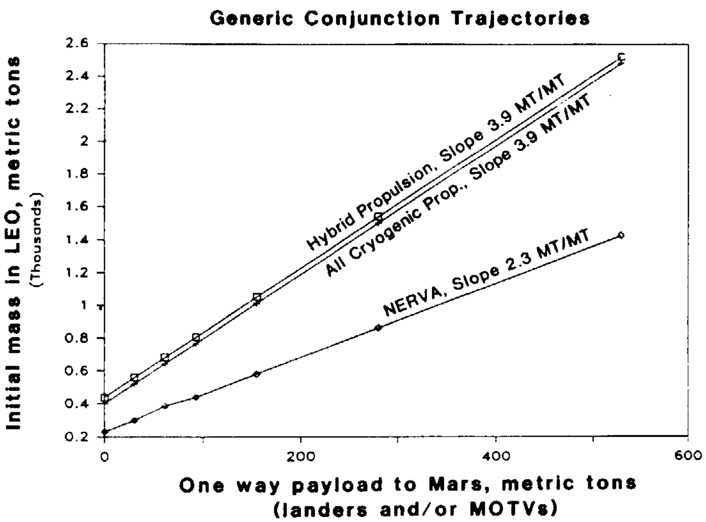
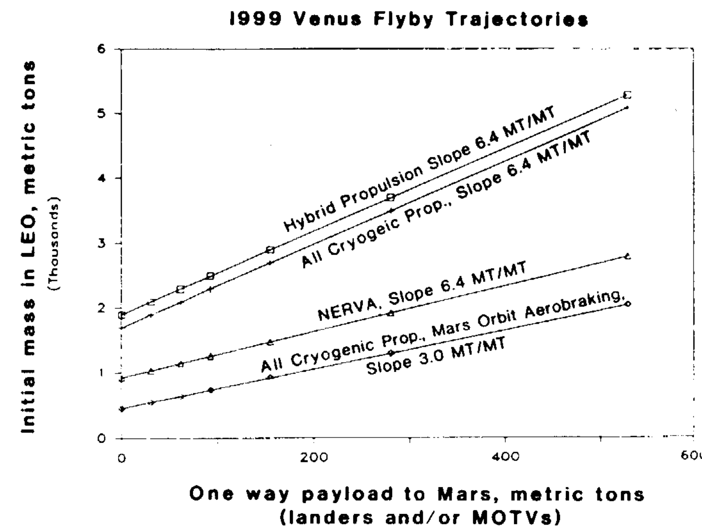
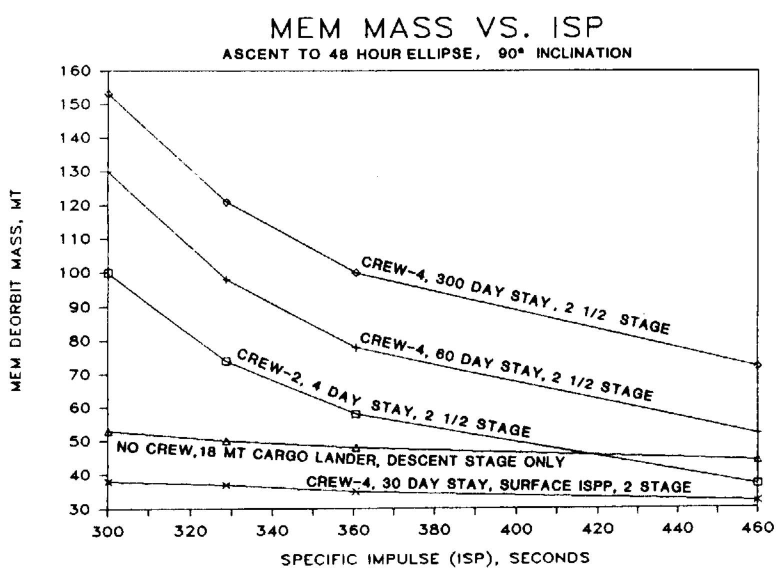


Fig. 13



All cases use a 53 metric ton Mission Module:
All cases are 3 stage, except last stage does TEI and EOI. Hybrid-first two stages are all cryogenic (H2/O2), last stage-O2/Propane.

Fig. 14



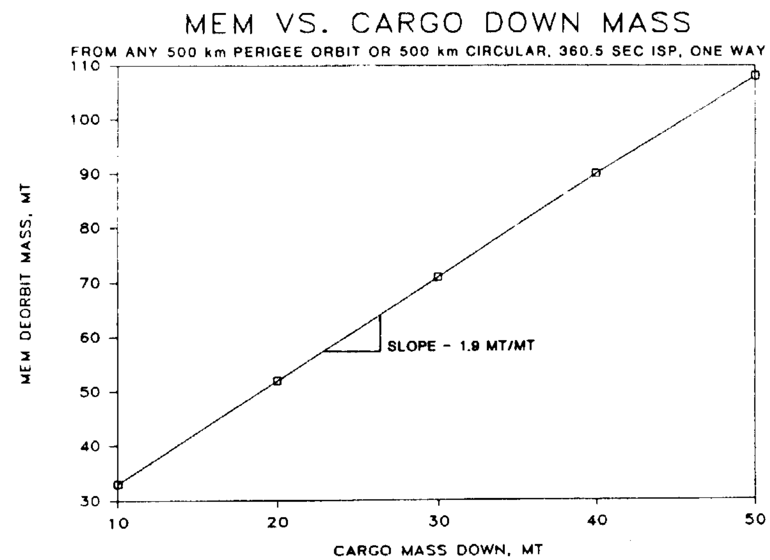


Figure 15

Cargo lander on the Lunar Surface

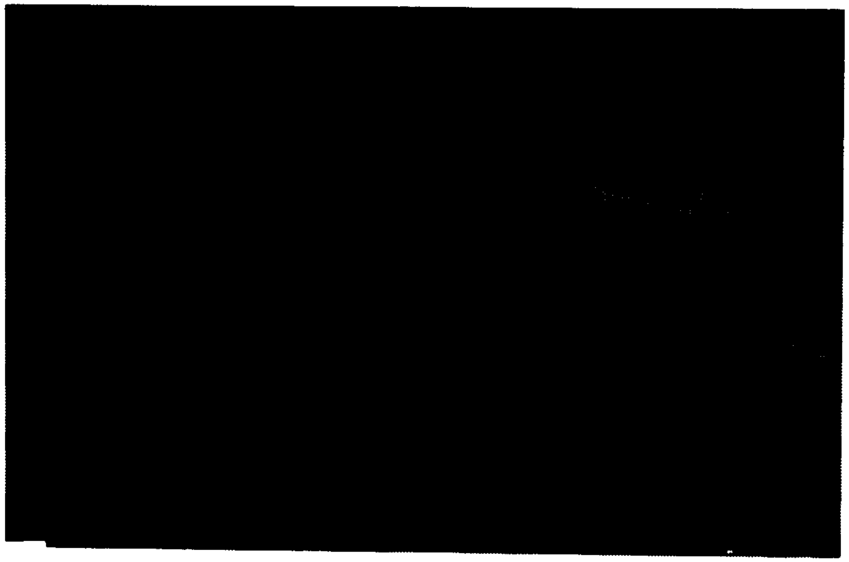


Figure 16

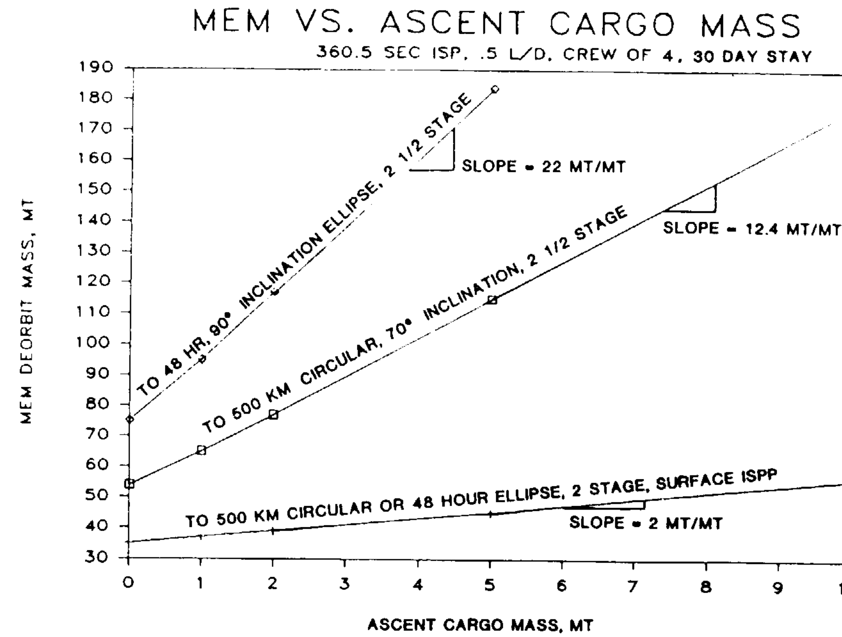


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Table 1

MEM DELTA V TABLE

MM ORBIT	HYPERBOLIC (EARTH RETURN) km/sec	60 HOUR 300 x 66,900 km km/sec	48 HOUR 500 x 57,000 km km/sec	24 HOUR 500 x 32,963 km km/sec	DEIMOS 20,160 km circular km/sec	PHOBOS 6,060 km circular km/sec	LOW CIRC. 500 km circular km/sec
DEORBIT DELTA V	0.2	0.2	0.2	0.2	0.667	0.574	0.2
LANDING DELTA Vs							
*BRAKE (Max. delta V, no chute)	0.92	0.92	0.92	0.92	0.92	0.92	0.92
*WIND (52 m/sec)	0.08	0.08	0.08	0.08	0.08	0.08	0.08
*HOVER (1 min., 1 nm range)	0.23	0.23	0.23	0.23	0.23	0.23	0.23
LANDING TOTAL*	1.23	1.23	1.23	1.23	1.23	1.23	1.23
ASCENT TO 185x93 km	3.9	3.9	3.9	3.9	3.9	3.9	3.9
DEST. ORB. INCLIN.	15	90	90	90	1.8	1.1	70
INCLIN. EFFECT	0.008	0.242	0.242	0.242	.000	.000	0.159
DRAG LOSS	0.15	0.15	0.15	0.15	0.15	0.15	0.15
CIRC. AT 185 km	0.02	0.02	0.02	0.02	0.02	0.02	0.02
TRANSFER TO FINAL ORBIT	2.405	1.32	1.3	1.226	1.75	1.257	0.17
ASCENT SUBTOTAL	6.48	5.63	5.61	5.54	5.82	5.33	4.40
10% CONTINGENCY	0.65	0.56	0.56	0.55	0.58	0.53	0.44
ASCENT TOTAL	7.13	6.20	6.17	6.09	6.40	5.86	4.84
TRANSFER FROM 500 KM CIRCULAR TO	2.38	1.25	1.23	1.14	1.66	1.13	-

km x .54 = nm   km x 3280 = ft

Table 2

MEM WEIGHT STATEMENT SUMMARY

MEM OPTION	MIN. MEM	30 DAY	60 DAY	300 DAY	CARGO MEM	SURFACE ISPP MEM, 2 STGE	REUSABLE MEM (SING. STAGE)
CREW	2	4	4	4	0	4	4
LIFE SUP. DURATION (days)	4	30	60	300	0	30	30
AERO L/D	0.5	0.5	0.5	0.5	0.5	0.5	0.5
ASCEND TO	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)	500x32963km (24 hour)
PRIMARY PROPULSION Isp, sec	360.5	360.5	360.5	360.5	360.5	360.5	460
LENGTH, m	8	8	8	8	8	8	-
DIAMETER, m	9	9	9	9	9	9	-
STORM SHELTER ?	NO	YES	YES	YES	NO	YES	NO
LANDED CARGO (MT)	1.91	1.91	1.91	1.91	18.00	1.91	1.91
ASCENT CARGO (MT)	0.14	0.14	0.14	0.14	0.00	0.14	2.05
DEORBIT MASS (MT)	56.52	73.71	76.38	98.30	48.31	35.09	451.05

TABLE 3

MEM WEIGHT STATEMENT

ASCENT TO 24 HOUR, 500 KM PERIAPSIS ELLIPSE.

MEM OPTION	MIN. MEM	30 DAY	60 DAY	300 DAY	CARGO MEM	SURFACE ISPP MEM, 2 STGE	REUSABLE MEM (SING. STAGE)
(ALL MASSES IN KGMS UNLESS OTHERWISE NOTED)							
ASCENT CAPSULE							
PRIMARY STRUCTURE	255	255	255	255	255	255	510
COUCH, RESTRAINTS	18	36	36	36	0	36	36
HATCHES, WINDOWS	55	55	55	55	55	55	55
DOCKING PROVISIONS	77	77	77	77	77	77	77
PANELS, SUPPORTS	23	23	23	23	23	23	23
BATTERY	123	123	123	123	123	123	123
EPS DISTRIBUTION	105	105	105	105	105	105	105
COMMUNICATIONS	95	95	95	95	95	95	95
GUIDANCE AND NAV.	102	102	102	102	102	102	102
CONTROLS & DISPLAYS	91	91	91	91	0	91	91
INSTRUMENTATION	86	86	86	86	0	86	86
LIFE SUPPORT SYS.	236	432	432	432	0	432	432
RCS - DRY	107	133	133	133	0	133	151
RCS - PROPELLANT	89	110	110	110	0	110	125
RETURN PAYLOAD	136	136	136	136	0	136	136
CREW	159	318	318	318	0	318	318
CONTINGENCY	195	242	242	242	93	242	274
ASCENT CAPSULE TOTAL	1,953	2,419	2,419	2,419	928	2,419	2,738
ASCENT PROPULSION							
STAGE 2 DELTA V, km/sec	2.66	2.66	2.66	2.66	0.00	0.00	0.00
TANK MASS/PROP. MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.07
2ND STAGE ISP, sec	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	460 (LO2/H2)
2ND STAGE MASS RATIO	2.12	2.12	2.12	2.12	1.00	1.00	1.00
TANKS & SYSTEM	243	294	294	294	0	304	0
ENGINE & INSTAL.	253	253	253	253	0	253	0
CONTINGENCY	50	55	55	55	0	56	0
BOILOFF & ULLAGE	316	382	382	382	0	0	0
USABLE 2ND STGE PROP	3,162	3,823	3,823	3,823	0	0	0
2ND STAGE PROP. WITH BOILOFF & ULLAGE	3,478	4,205	4,205	4,205	0	0	0
2ND STAGE PROPULSION SYSTEM MASS TOTAL	4,025	4,807	4,807	4,807	0	613	0
2ND STAGE IGNITION MASS	5,978	7,226	7,226	7,226	928	3,032	2,738
1ST STAGE DELTA V km/sec	3.43	3.43	3.43	3.43	0.00	0.00	0.00
TANK MASS/PROP. MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.07
1ST STAGE ISP, sec	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	460 (LO2/H2)
1ST STAGE MASS RATIO	2.64	2.64	2.64	2.64	1.00	1.00	1.00
TANKS & SYSTEM	1,083	1,309	1,309	1,309	0	1,382	0
ENGINE & INSTAL.	0	0	0	0	0	0	0
CONTINGENCY	108	131	131	131	0	138	0
BOILOFF & ULLAGE	1,407	1,700	1,700	1,700	0	0	0
USABLE 1ST STGE PROP	14,066	17,004	17,004	17,004	0	0	0
1ST STAGE PROP. WITH BOILOFF & ULLAGE	15,473	18,704	18,704	18,704	0	0	0
1ST STAGE PROPULSION SYSTEM MASS, TOTAL	16,664	20,144	20,144	20,144	0	1,520	0
1ST STAGE IGNITION MASS (TOT. ASCENT)	22,642	27,370	27,370	27,370	928	4,552	2,738

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TABLE 3

MEM WEIGHT STATEMENT (CONT'D.)

MEM OPTION	MIN. MEM	30 DAY	60 DAY	300 DAY	CARGO MEM	SURFACE ISPP MEM, 2 STGE	REUSABLE MEM (SING. STAGE)
DESCENT STAGE							
JETTISONED STRUCTURE	2,114	2,114	2,114	2,114	2,114	2,114	0
RETAINED STRUCTURE	2,477	2,477	2,477	2,477	2,477	2,477	7,500
SEC. STRUCTURE	409	409	409	409	409	409	409
LAB STRUCTURE	477	3,810	3,810	3,810	0	3,810	477
ELECTRICAL PWR SYS.	253 (2kw fcell)	1,009 (2kw fcell)	1,882 (2kw fcell)	8,864 (2kw fcell)	0	1,009 (2kw fcell)	1,009 (2kw fcell)
POWER DISTRIBUTION	182	182	182	182	0	182	182
COMMUNICATION	168	168	168	168	0	168	168
GUIDANCE & NAV.	5	5	5	5	0	5	5
CONTROLS & DISPLAYS	5	5	5	5	0	5	5
INSTRUMENTATION	114	114	114	114	114	114	114
LIFE SUPPORT SYS. (open loop)	22 (2kw fcell)	621 (2kw fcell)	1,169 (2kw fcell)	5,555 (2kw fcell)	0	621 (2kw fcell)	621 (2kw fcell)
RCS - DRY	441	575	596	767	376	273	3,613
RCS - PROPELLANT	912	1,191	1,234	1,588	780	566	7,484
LANDING GEAR	991	991	991	991	991	991	991
NET LANDED PAYLOAD	1,909	1,909	1,909	1,909	18,000	1,909	1,909
CONTINGENCY	1,164	1,731	1,896	3,217	2,807	1,628	10,494
DESCENT SUBTOTAL	11,643	17,310	18,960	32,175	28,068	16,281	34,981
DESCENT PROPULSION							
DESCENT DELTA V, km/sec	1.23	1.23	1.23	1.23	1.23	1.23	7.32
TANK MASS/PROP. MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.06
DES. STAGE ISP, sec	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	360.5 (LO2/MMH)	460 (LO2/H2)
DES. STGE MASS RATIO	1.42	1.42	1.42	1.42	1.42	1.42	5.07
TANKS & SYSTEM	1,144	1,493	1,547	1,991	978	710	21,961
ENGINE & INSTAL.	504	704	704	1,000	704	704	2,000
CONTINGENCY	165	220	225	299	168	141	2,396
BOILOFF & ULLAGE	925	1,207	1,251	1,610	790	574	20,718
USABLE DES STGE PROP	15,418	20,116	20,847	26,839	13,175	9,563	345,304
DES. STGE PROP. WITH BOILOFF & ULLAGE	16,344	21,323	22,097	28,449	13,965	10,136	366,022
DESCENT STAGE PROPULSION MASS	18,156	23,740	24,573	31,740	15,815	11,691	392,379
DES. STAGE IGNITION MASS (ENTRY MASS)	52,442	68,420	70,904	91,285	44,811	32,524	430,100
DEORBIT PROPULSION							
DEORBIT DELTA V, km/sec	0.20	0.20	0.20	0.20	0.20	0.20	0.20
DEOR. TANK/PROP MASS	0.07	0.07	0.07	0.07	0.07	0.07	0.06
DEORBIT ISP, sec	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	300 (GOOD SOLID)	460 (LO2/H2)
DEORBIT MASS RATIO	1.07	1.07	1.07	1.07	1.07	1.07	1.05
TANKS & SYSTEM	260	339	352	453	222	162	1,174
ENGINE & INSTAL.	100	100	100	100	100	100	200
CONTINGENCY	0	0	0	0	0	0	0
BOILOFF & ULLAGE	0	0	0	0	0	0	0
USABLE DEORBIT PROP	3,717	4,847	5,023	6,465	3,177	2,308	19,574
DEORBIT PROP. WITH BOILOFF & ULLAGE	3,717	4,847	5,023	6,465	3,177	2,308	19,574
DEORBIT STAGE	4,077	5,287	5,475	7,017	3,500	2,569	20,948
DEORBIT IGNITION MASS (MEM TOT. MASS)	56,519	73,707	76,378	98,302	48,310	35,094	451,048

Table 3 and the plots use the basic Rockwell design, first stage descent and second stage ascent concepts with drop tanks, and an open loop life support system, using 2 KW fuel cell power. No life support volume calculations were performed. No chutes or ballutes were included. 10% ascent delta V and 10% dry mass contingency numbers were used. A 3.3 metric ton storm shelter for solar flares was used for all configurations except the four day stay and reusable, single stage MEM. Boiloff was limited to 10% of usable stage propellant for the ascent stages. This assumption may not be realistic for the longer surface stays.

Seven different vehicle designs were addressed: (1) A minimum MEM (4 day stay for a crew of two), (2) 30 day stay MEM, (3) 60 day stay MEM, (4) 300 day stay MEM, (5) A cargo lander, (6) Surface-produced-propellant using MEM (in situ propellant production, or ISPP), and (7) A reusable single stage MEM. Table 2 summarizes their characteristics for one case for which a weight statement (Table 3) is included.

The single stage reusable MEM numbers in the tables should be viewed with caution because they are a distant extrapolation from the original Rockwell vehicle. All structural mass was doubled, and a 30% contingency on dry mass was added (up from 10%). Iterative calculations assuming two metric tons payload up and down plus a crew of four and 30 days consumables resulted in the following numbers for a single stage reusable MEM:

Case		Mars Entry Mass
To a 60 hour ellipse, 360.5 sec. Isp	-	1,206 m. tons
To 500 km circular, 360.5 sec. Isp	-	300 m. tons
To 500 km circular, 460 sec. Isp	-	157 m. tons
Surface ISPP for ascent stage only, 300 sec. Isp, to any orbit	-	83 m. tons
Surface ISPP for ascent stage only, 460 sec. Isp, to any orbit	-	69 m. tons

At least in terms of simple mass calculations, a single stage reusable MEM does not appear to be out of reason. A substantial infrastructure in Mars orbit or on the surface will be needed to maintain it, however.

REFERENCES

1. Canetti, G. S., Definition of Experimental Tests for a Manned Mars Excursion Module, three volumes, Contract # NAS9-6464, North American Rockwell Corp., Columbus, Ohio.

   Volume 1 - Summary Final Report, Oct. 1966-Aug. 1967, 1968, 78 pages, 68X12107*#, Report # NASA-CR-65911 SD-67-755-1.

   Volume 2 - Design Final Report, 1968, 571 pages, 68X11978, Report # NASA-CR-65912 SD-67-755-2.

   Volume 3 - Test Program Final Report, 1968, 256 pages, 68X12010*, Report # NASA-CR-65913 SD-67-755-3.

2. Woodcock, Gordon R., and Vinopal, Timothy J., Transportation Modes for Manned Mars Missions, Boeing Aerospace Company.

3. Impact of Lunar and Planetary Missions on the Space Station, Eagle Engineering Final Report, Report # 84-85D, Contract # NAS9-17176.

4. Case for Mars II Viewgraphs, The Planetary Society, University of Colorado and Boulder Center for Science and Policy. Copyrighted sketch by Carter Emmart.