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**A MANNED MARS MISSION CONCEPT WITH ARTIFICIAL GRAVITY**

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**ABSTRACT**

A series of simulated manned Mars missions was analyzed by a computer model developed by the author under contract to NASA - JSC. Numerous mission opportunities and mission modes were investigated. Sensitivity trade studies were performed of the vehicle all-up mass and propulsion stage sizes as a function of various levels of conservatism in mission velocity increment margins, payload mass and propulsive stage characteristics. This study emphasized the longer duration but less energetic type of conjunction class mission. The specific mission opportunity reviewed was for a 1997 departure.

From the trade study results (some 300 separate mission simulations), a three and one-half stage vehicle concept evolved, utilizing a Trans-Mars Injection (TMI) first stage derived from the Space Shuttle External Tank (ET). The vehicle was completely ground assembled but required propulsion system reconfiguration, refueling with liquid hydrogen and oxygen, and payload mounting in Low Earth Orbit (LEO), utilizing the services available from the LEO Space Station. The second stage, used solely for propulsive Mars Orbit Insertion (MOI) of the space vehicle into a 24 hour period orbit about Mars, utilized cryogenic propellants ( $O_2/H_2$ ) and advanced, active thermal control features to preserve the liquid hydrogen over the 9 month duration journey from Earth orbit to Mars. The final "stage and one-half" propellants chosen were liquid oxygen and liquid propane in recognition of the formidable problem of retaining liquid hydrogen during the 15 month loiter in the vicinity of the heat-emitting planet Mars. Following the "Trans-Earth Insertion (TEI)" burn, the spent TEI propellant tanks were jettisoned and the remaining tankage provided for propulsive return of the "Command Module (CM)" with the crew into a 24 hour elliptical orbit of Earth. Final crew recovery was accomplished by a man-rated Orbit Transfer Vehicle (OTV).

Artificial gravity was provided for 40 metric tons of living quarters in two "Mission Modules (MMs)" mounted on outrigger tunnels extended from the spinning central core which contained a heavily-shielded, de-spun CM of 40 metric tons mass serving as both command station for the

mission and "storm shelter" to allow survival of the crew during solar flare events.

Two 75 metric ton landers were provided, permitting crews of four persons two surface stays of two to four weeks duration each at separate locations on the surface of Mars. Four "Mars Maneuvering Vehicles (MMVs)" were also provided to permit manned sorties from the Mars orbit to both Phobos and Deimos during the Mars orbit staytime. The MMVs were also employed to recover the ascent stages of the two landers from low (500 km) circular Mars orbits.

The aggregate payload mass for this mission was 287 metric tons and the departure mass from Earth orbit was 1,254 metric tons, over 60% liquid oxygen. Generous electrical power service was provided for the mission by a cluster of "SP-100" class nuclear-electric generators energized after the TMI burn and left in solar orbit at mission's end.

#### TRAJECTORY SELECTION

A round trip mission from Earth orbit to the planet Mars is not necessarily more energetic than other missions contemplated as precursor activities. A round trip mission from Low Earth Orbit (LEO) to the geostationary Earth orbit requires a total propulsive velocity change of about 8.5 km/sec. A round trip journey to the Earth's moon requires about 5.5 km/sec if aerobraking is used for Earth return as was done on the Apollo missions. The "Conjunction Class" missions to Mars described in this paper require 6 to 7 km/sec, not including mission reserves. The mission would return to a high elliptical Earth orbit accessible from the Space Station by means of a man-rated OTV. The four major maneuvers--Trans-Mars Insertion, Mars Orbit Insertion, Trans-Earth Insertion and Earth Orbit Insertion (EOI)--of the mission are relatively invariant with the specific bi-annual mission opportunity used for the mission, permitting a single design to serve a series of mission opportunities.

If low energy "Hohmann transfer" heliocentric orbits are employed, the time of flight is seven or more months each way. Awaiting favorable alignment of the planets requires stay times in orbit about Mars on the order of 15 months. Total elapsed time for the Conjunction Class missions is therefore over 2 1/2 years.

Many schemes have been proposed to reduce these flight times. However, each scheme has drawbacks, mainly, a large increase in required

mission energy. Use of a Venus Swingby approach is one means of reducing this mission energy penalty. This strategy brings the interplanetary vehicle much closer to the Sun than do the Hohmann transfers and poses thermal management problems that must be solved.

The "Opposition Class - Venus Swingby" missions discussed in the literature have total mission durations of about 23 months-- with limited time (1 to 2 months) available to orbit or land on Mars and accomplish the mission before the return trip to Earth. Mission energy ranges from 9 to 14 km/sec and is highly variable in the magnitude of its four components, each being highly dependent upon the specific mission opportunity selected.

The type of mission chosen is driven by the relative importance of mission energy magnitude and variability versus mission elapsed time. In the era of "permanent human habitation of space", the 30% increase in mission duration of the Conjunction Class mission is of lesser importance than in the Apollo era, when most of the Mars literature was written. Rather than being a detriment, the long stay time at Mars of the Conjunction Class mission can be productive and increase the science return of the mission.

Previous space transportation studies (NASA-Boeing FSTSA, 1976, NASA-DOE SPS, 1977) indicate that the cost of launching the vehicle components and propellants into Low Earth Orbit dominate the total mission cost for advanced space missions. A good indicator of a Mars mission cost is thus the "Initial Weight in Low Earth Orbit (IWLEO).

Sensitivity studies presented at this conference show that the Conjunction Class mission can deliver four times the useful payload round trip Earth-Mars-Earth or six times the payload from Earth orbit to orbit around Mars than can the Opposition Class mission for a given propulsion technology and IWLEO. These factors may be different for advanced propulsion technologies or with the development of "in-situ propellant production", but the trend remains - a large increase in payload mass is available to the Conjunction missions compared to the Opposition, even Swingby, missions. This added payload capability will allow for more experimentation equipment to be used during the mission's extended time in the vicinity of Mars.

Increased payload capability can also be used to enhance crew comfort. The requirement to spend more than two years in space, remote from the conveniences and companionship of Earth means more attention must be paid to human needs. No degree of training and motivation will allow a crew member to endure this long without the ability to briefly get away from others, to be frequently stimulated by interesting events, to have a change of environment and to enjoy a diet and personal services more Earth-like than heretofore provided to astronauts. For this reason, it is suggested that each mission carry two Mars landing vehicles and the vehicular capability to visit the moons of Mars during the long stay time at Mars. It is also suggested that multi-compartment living quarters, with artificial gravity and separated from the workplace, are required even though these conveniences may not be clearly necessary from purely physiological considerations.

For these reasons, a large space vehicle, aggregating some 1250 metric tons and carrying 287 metric tons of useful payload is proposed to accomplish one or a series of missions. The intent of this mission concept is to provide a point of departure and framework for future discussion and study.

#### PRIMARY PROPULSION (TMI) SYSTEM OPTIONS

The manned Mars mission was a feature of the post-Apollo "Integrated Plan" for proposed future NASA space activities. At that time, about 1970, it was envisioned that Saturn-derived launch vehicles, the "Int-20 and Int 21" would continue to be launched immediately following the Apollo program and later co-exist with a vehicle similar to but smaller than today's Space Shuttle to provide crew launch services. With a capacity of almost 110 metric tons per launch (compared to the Shuttle's 29.5 metric tons), the Int-21 represented a powerful instrument for proposed space activities.

A large, 10 m diameter Space Station was envisioned in Low Earth Orbit as the home base for a man-rated, multi-purpose "Space Tug" using hydrogen/oxygen propellants at a specific impulse of 460 seconds and a "Reusable Nuclear Shuttle (RNS)" for ferry missions to lunar orbit and to Mars. The RNS was to use the "NERVA" thermal reactor power plant heating 136 metric tons of liquid hydrogen to provide thrust of up to 1/3 megaNewton (75,000 lbf) at a peak specific impulse of 800 seconds.

The Tug was to have modular add-ons to permit landing on the Moon and to carry humans to the Geostationary Orbit and return.

The NERVA engine and the M-1 engine, a gas generator cycle hydrogen/oxygen engine rated at 4.5 MN (1,000,000 lbf) enjoyed vigorous development activity. None of these plans reached fruition.

Aside from the magnificent Space Shuttle Main Engine (SSME) and modest improvements in bi-propellant liquid propellant engines, little of real significance has been accomplished in high thrust space propulsion in the intervening 15 years. A "Spacecraft Propulsion Systems" seminar held by the AIAA in May 1984 predicted that, with concentrated effort, improvements of 4 orders of magnitude in specific impulse might be attained in the next 25 years, culminating with anti-matter propulsion at a specific impulse of 1,000,000 seconds by 2010 or so. Although such developments may possibly occur and would assuredly enable space travel for the masses, recent history and the dearth of true breakthroughs do not lend high confidence to these optimistic forecasts.

The U.S. DOD "Strategic Defense Initiative (SDI)" may revitalize this field and produce, for civil use, new and markedly improved space propulsion systems. For planning circa 2000 manned Mars missions, however, only two high thrust propulsion technologies now appear to be available that quickly - chemical propulsion, as exemplified by the SSME, and nuclear thermal propulsion of the NERVA type. An assortment of other, smaller bi-propellant liquid rocket engines will be needed to achieve the total mission objectives which can have small but cumulatively significant improvements in performance when compared to the rocket engines available today. Given the 6 to 12 year gestation interval for new space propulsion elements, serious development must soon begin in order to meet the turn of the century goal for the manned Mars mission.

Since the 2.2 MN (500,000 lbf) thrust class SSME, as applied to the Shuttle, must safely operate at sea level atmospheric pressure, the extent to which the exhaust gases can be expanded in the nozzle is necessarily constrained. If future variants of the SSME are intended to only be used in the space environment, this constraint is removed, and a larger bell nozzle can be used to improve the SSME performance to the 470 second specific impulse range. With an inert mass of less than 4 tons, a

high expansion SSME is an attractive choice for the first and most difficult maneuver of the mission - Trans-Mars Insertion.

Its competitor is a latter-day variant of the NERVA engine. Improvements in the carbon core material of NERVA have been proposed which might permit specific impulse levels as high as 850 seconds. This type of engine, with the necessary shadow shield to protect the crew from the radiation field of the engine, is inherently massive as compared to combustion engines of higher thrust such as the SSME. This inert mass, coupled with the high inert mass of the hydrogen fuel storage vessels, degrades the apparent advantage over the SSME type engine. When operational constraints and penalties due to low thrust and the radiation field and safety measures required to protect both the space operations and the population of Earth are accommodated, the apparent advantages further decline. If used at all, a nuclear engine must be initially energized or stored for reuse only at "nuclear safe" orbital altitude (defined as that altitude which would permit natural decay to safe levels of the radioactive products generated within the engine during its operation, before atmospheric drag would cause re-entry of the vehicle). This constraint could force final assembly of the Mars space vehicle to take place at altitudes sufficient to induce another radiation problem - encounter with the intense natural radiation field of the trapped radiation belts around Earth.

Future studies will be needed before the relative merits of the two competing propulsion technologies are fully understood. Vital to the trade study is the cost of placing propellants into Low Earth Orbit. Therefore, such a study must consider not only the technical and operational factors described above but also the characteristics of the launch capability available for use. Monetary tradeoffs must also have available a reliable "mission model", as apparent savings in mission costs may be overwhelmed by disparities in development cost between the alternative space vehicles and their supporting infrastructure. Thus, an indefinite series of missions can better justify a larger development cost for primary propulsion systems than can a single mission or a short series of missions.

## PROPULSIVE VEHICLE STAGING

The mission simulation routine used for this analysis was set up to permit the use of two propulsive stages for each of the major maneuvers, or "burns", including perigee raise in Earth orbit, to permit the computational routine to be employed for "aerobraking" upon return to Earth. Since the Conjunction Class of mission was chosen as the "baseline" for this study, the velocity increments for each of the four major burns were not sufficient to justify two stages for any of them.

For other reasons, staging was found to be called for just prior to initiation of each of the last three major burns. The TMI stage carried 716 tons of oxygen/hydrogen propellants (57% of total space vehicle mass) at an Oxidizer-to-Fuel (O/F) ratio of 6.0:1 and was powered by a single engine derived from the SSME which delivered 468 seconds of specific impulse. This stage performs a single burn of about 25 minutes duration within a few hours of separation from the Low Earth Orbit Space Station (LEO SS) and has then completed its principal function. Thrust-to-mass ratio is 0.17 at ignition and 0.40 at burnout. Inert mass of this stage is estimated by two independent methods to be about 43 tons. As a consequence of its short mission life, the cryogenic insulation provisions necessary to retain propellant servicing at the LEO SS are expected to be fully adequate for the flight mission. To provide shielding from natural radiation and to permit, if desired, use of residual fluids by the spacecraft, the spent TMI stage is retained until Mars approach.

To acquire the 24 hour period orbit at Mars, the second stage of the space vehicle is used. This stage also uses oxygen/hydrogen propellants, carries 102 tons of propellant in heavily insulated and actively refrigerated tanks and produces a 0.20 thrust-to-mass ratio at ignition by use of a 920 kN (207,000 lbf) thrust engine of new design with high chamber pressure and an ultra-high expansion ratio nozzle to deliver 480 seconds of specific impulse. Smaller, multiple engines may be found to be preferable on further analysis. Including the insulation and active refrigeration provisions, this stage has an estimated inert mass of 18 tons. As the outbound mission flight time is in the vicinity of nine months, propellant conditioning technology will be challenged to minimize or eliminate hydrogen loss through tank venting. It is expected that this can be done by the time this mission is to be dispatched. This stage is

assumed to provide a 100 m/sec orbit adjust burn shortly after arrival in Mars orbit. The spent MOI stage is retained until shortly before the space vehicle is to begin the return to Earth.

Since the vehicle remains in orbit around Mars for about 15 months and is subjected to a higher heat load there than in free space due to the albedo of the planet, use of liquid hydrogen as the Earth return propulsion fuel is not considered to be a likely nor prudent choice. Instead, the higher boiling point fuel, liquid propane, is selected to be burned with oxygen for this third stage of the Mars space vehicle.

Oxygen and propane have an overlapping liquid range and therefore do not require thermal isolation from one another. Two propellant tanks with a capacity of 36.5 metric tons supply propellant to begin the return flight. Multiple engines (three?) of new design, with a total thrust of 300 kN (65,000 lbf) produce a thrust-to-mass ratio of 0.20 and deliver 373 seconds of specific impulse over a burn interval of a bit less than 8 minutes. Spent mass associated with the TEI burn totals 3.6 metric tons. The engines are retained for later use in acquiring Earth orbit.

Reuse of the TEI propulsion system occurs following the approximately seven month trans-Earth coast to acquire a highly elliptical 24 hour period Earth orbit in the orbital plane of the LEO SS. Only the central Command Module is present as the payload for this maneuver - the artificial "g" living quarters, stores modules, nuclear power supply and supporting structure are staged and left in the heliocentric Earth-Mars orbit. Approximately 19.5 metric tons of propellant are required, stored in partly-filled tanks of the same design as those used for TEI.

Inert mass of this propulsion system is estimated to be 2.4 tons. A thrust-to-mass ratio of 0.70 is experienced near end of burn, unless fewer than three engines ignite or the engines are throttled to lower than full rated thrust. A single engine of the cluster of three is adequate to safely complete this final maneuver.

Secondary propulsion systems of the Mars space vehicle include a pair of RL-10 engines on the large TMI stage to permit roll control, a gas oxygen/hydrogen reaction control system to provide 100 m/sec mid-course correction and attitude change during trans-Mars flight. About 12.5 metric tons of cryogenic propellant are used at a specific impulse of 400 seconds for these maneuvers. These propellants can be tank vent



gas from the MOI stage, relieving somewhat the thermal control difficulty. Inert mass of this RCS is estimated to be 8.2 metric tons.

A separate oxygen/propane RCS is used for the 100 m/sec of maneuvers assumed necessary during inbound flight. 3.5 tons of oxygen/propane propellant are consumed at a specific impulse of 315 seconds by an RCS of 2.27 tons inert mass which is integral with the Command Module.

Many other combinations of staging arrangement and propellant selection for the manned Mars missions are possible. One or more combinations different from those described above may be found, on more detailed analysis, to be preferred. The ones described above were selected with operational suitability in mind and other candidate systems will have to provide at least equal attention to operability as well as performance. Electric propulsion is a provocative alternative for at least some of the propulsive functions, and deserves more attention if a power-rich environment may be economically provided.

#### HABITATION CONSIDERATIONS

Several important principals need to be observed in design of the manned Mars mission and its space vehicle. First, there must be sufficient human resources and skills present for this microcosm of society to be entirely self-sufficient (except for information) for the two to two and one-half year journey. This will require bakers and barbers as well as planetary scientists and pilots. New and unexpected maintenance and repair tasks will require tools, supplies, and information adequate to the potential tasks. Equally unexpected science opportunities are apt to present themselves for exploitation. What this indicates regarding minimum crew size remains to be determined. It is likely that the estimated minimum crew will increase rather than decrease with time as the studies unfold. Perhaps design accommodations for 12 to 20 persons is not excessively conservative if 6 to 8 are planned today.

A favorable mix of fully committed personalities will also be necessary, as, even if spacious by contemporary spacecraft standards, the Mars vehicle will become a confining and overly intimate place before the mission ends. A key to rendering this close proximity of people tolerable may be to permit interruption of their intimacy. The space vehicle should have the workplace and living quarters sufficiently apart to require a conscious "going to work" personal journey from private

quarters each day. The Mars excursion modules should be pressurized and, although a bit remote from the normal living and working quarters, should be accessible via pressurized compartments so that periodic subsystem checks may be periodically performed - the principal beneficiary being perhaps the human subsystem.

While in orbit around Mars, the excitement and newness of personal, on-site science experimentation will no doubt assure complete harmony among the entire crew. If there can be, on each Mars mission, two landings on the surface and visitation to both of the Martian moons, the crew will always have either a task underway, data to reduce or planning to review for the next critical event. Care should be taken that useful work is reserved or provided for the 9 month return journey.

One final consideration relates to the provisioning of a heavily shielded "storm shelter" for the crew to gather in during solar flare events. This place must be occupied for the duration of the atypically high natural radiation levels which may persist several days or possibly weeks. This close confinement may not come at all during a mission or it may be repeated. Rather than making a very small volume dedicated to this purpose, which requires total inactivity of the occupants, it will prove more acceptable to the crew to arrange provisions, including propellants, around the Command Module so that, even though normal living quarters are inaccessible during these intervals, useful work continues. If necessary, additional shielding mass beyond theoretical minima should be provided to assure that adequate volume for near-normal activity is maintained during these stays.

These considerations, coupled with the need for an artificial gravity field previously discussed, have led to the suggested general arrangement of an artificial gravity manned Mars space vehicle illustrated by Figures 1 through 4. As these design matters will continue to be highly subjective, different people will have different and strongly defended views as to what is "correct" for the mission.

Throughout these discussions, it must be remembered what people on this journey are asked to do--remain in close quarters functioning at peak efficiency with no possibility of altering their circumstance for a very long time.

**MANNED MARS MISSION**  
 CONJUNCTION CLASS  
 NEAR TERM TECHNOLOGY OPTION  
 HYBRID CHEMICAL PROPULSION  
 1400 METRIC TON CLASS

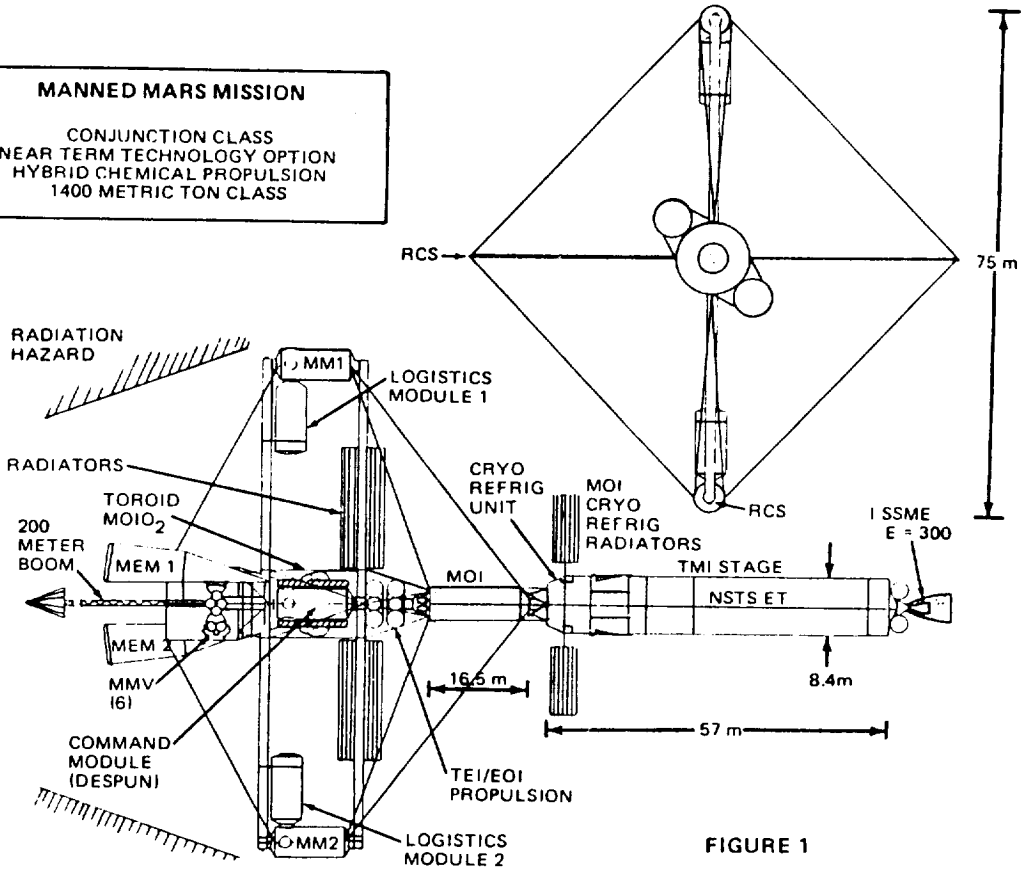


FIGURE 1

INITIAL MARS ORBIT CONFIGURATION

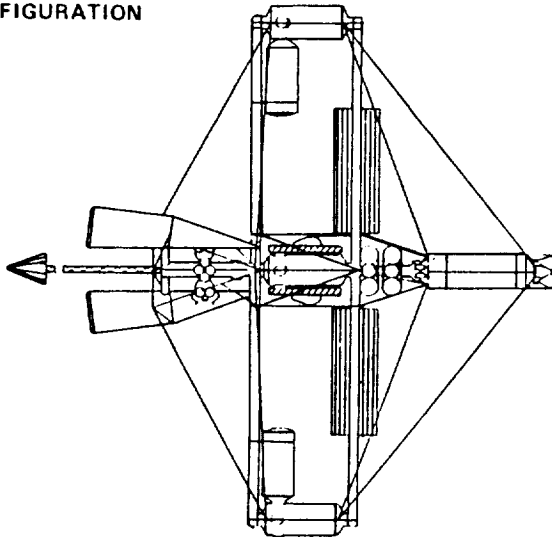


FIGURE 2

TRANS-EARTH CONFIGURATION

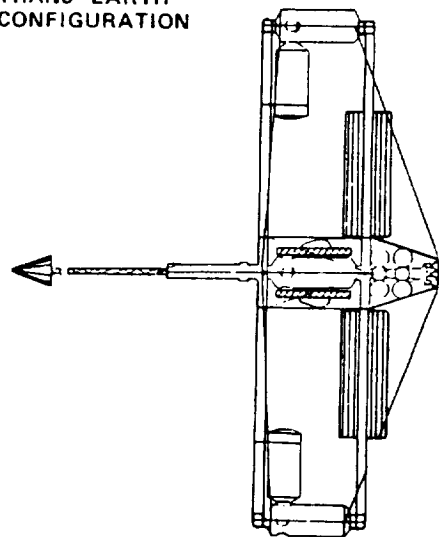


FIGURE 3

FIGURE 4



EARTH ORBIT CONFIGURATION

## CANDIDATE SPACE VEHICLE GENERAL ARRANGEMENT

The general arrangement of the candidate manned Mars mission space vehicle in the Earth departure configuration is illustrated by Figure 1. It is a three and one-half stage propulsion vehicle system with a large spacecraft "cluster" aggregating almost 300 metric tons mounted forward of the three stages. The spacecraft cluster is comprised of: (a) Two "Mission Modules (MM's)", which serve as living quarters for the crew for the two and one-half year duration mission; (b) The core "Command Module (CM)" which serves the purposes of on-board command and control, a central meeting place for conferences, meals and recreation and for a radiation-hardened "storm shelter" for crew habitation and support during the abnormally high natural radiation levels occurring infrequently as a consequence of solar activity; (c) Two "Mars Excursion Modules (MEM's)" which accomplish the descent from the 24 hour period Mars orbit of the "mother ship", support the crew for their two week surface stay, and return the crew and mission artifacts to a 500 km altitude circular Mars orbit; (d) Four "Mars Maneuvering Vehicles (MV's) which provide mobility to the crew in the vicinity of the high Mars orbit, transport a "Crew Module" to soft land on both moons of Mars - Phobos and Deimos - and return to the "mother ship", and finally to recover the ascent stages/crew compartments of the MEMs from their 500 km circular orbits to the high ellipse; (e) A nuclear-electric power supply, made up of a cluster of "SP 100" or later nuclear reactor/thermal cycle power plants with the associated space radiator, control and power conditioning subsystems; (f) Four tunnels interconnecting the MM and CM; (g) An extensible boom interconnecting the CM and power supply; (h) Structural ties to lend rigidity to the spacecraft cluster; (i) Stage separation equipment to permit the spacecraft to shed all appendages from the CM just before the final propulsive maneuver into a 24 hour period Earth orbit.

Figure 2 illustrates the vehicle configuration as it appears just before insertion into Mars orbit and as it will remain during MEM and MMV operations during the 15 month stay in the Mars orbit. Overall dimensions of the departing space vehicle are approximately 120 m long and 75 m in platform span. The total mass at Earth departure is approximately 1250 metric tons.

The large "Trans-Mars Insertion (TMI) Stage" is jettisoned just prior to the Mars Orbit Insertion (MOI) burn. It remains with the space vehicle for the trans-Mars journey to provide radiation shielding mass and to permit possible recovery and use of the stage residual propellants and subsystems capabilities. In a similar fashion, the MOI stage remains with the space vehicle after it has performed its primary mission until preparations begin for return to Earth.

Figure 3 illustrates the post-TEI stage configuration which remains in this arrangement for the trans-Earth flight for the same reasons that the TMI stage was retained for the outbound leg. In both cases, a penalty is paid in attitude control and mid-course correction propellant in exchange for the utility these spent stages may possess. More detailed studies will be necessary to determine whether or not retention is an effective and economical choice.

In the last day of the mission before beginning the Earth Orbit Insertion (EOI) maneuver, the crew gathers in the CM with all of the science yield and mission documentation. The MM's, nuclear powerplant, tunnels and supporting structures are then jettisoned to remain in the heliocentric orbit. Post-jettison propulsion may be required for the larger masses, particularly the spent nuclear reactors, to assure no future contact with the biosphere of Earth or interference with future space missions. Batteries power the CM during the 12 to 36 hours necessary for recovery of either the entire CM or only the crew and science yield from the 24 hour period ellipse to the LEO Space Station. Future Orbit Transfer Vehicle (OTV) capabilities and mission costs will determine the recovery scenario. Figure 4 illustrates this final mission configuration. Dimensions are roughly the same as the Space Station "Common Modules", about 4.5 m diameter by 12 to 15 m in length, and mass is about 40 metric tons. Thus, only a bit over three percent of the mass dispatched from the LEO Space Station for this mission will be recovered.

#### STAGE TANKS

The mission simulation studies mentioned earlier consistently indicated TMI propellant quantity required in the 500 to 750 metric ton range. As the Space Shuttle External Tank (ET) has a capacity of 707 metric tons, it was elected to conceptualize a means of achieving Trans-Mars Insertion by using the ET as the propellant container for the TMI

stage. The TMI stage would thus be completely assembled before launch. It would be used as a standard STS ET for launch of a shuttle, carrying useful payload to the Space Station and placing the TMI stage into orbit. The "ET Stage" was therefore conceptually defined, utilizing a single modified SSME for primary propulsion. This general concept (the "OIS" studies by MSFC, RI, and MDAC) has been reviewed earlier by NASA MSFC and others, circa 1970.

The second stage of the space vehicle also utilizes Oxygen/Hydrogen propellants, but is a much smaller stage than the 1st stage - (requiring a propellant load available at arrival at Mars) of 102 metric tons.

The final stage serves the dual purposes of departing from Mars orbit and placing the CM into high Earth orbit to complete the mission. Since its first use is almost two years after launch, the less volatile fuel--liquid propane--was selected to ease the problem of boil-off loss or reliquefaction. The propellant is contained in eight spherical tanks, each less than 2.6 m in diameter. This stage may either be assembled in space or launched as a single Shuttle payload fully assembled and loaded with propellant in LEO.