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# MANNED MISSION TO MARS WITH PERIODIC REFUELING FROM ELECTRICALLY PROPELLED TANKERS

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## Abstract

In a joint study by students from the Ecole Polytechnique Fèminine, France, and the University of California, Los Angeles, a mission concept that had the objective of evaluating the feasibility of a non-nuclear, yet fast, manned mission to Mars was considered. Ion-engine propelled tankers are postulated that would provide mid-course refueling of LOX and $LH_2$ to the manned ship. The scenario is therefore one of a "split mission", yet with the added feature that the cargo ships include tankers for mid-course refueling. The present study is a continuation of one first conducted last year. Emphasis this year was on the design of the tanker fleet.

## Introduction

Ion engine and other electric thrusters can have a very high specific impulse, but, for realistic levels of electric power on a space vehicle, have low thrust, resulting in very long travel times. In this mission study, it is proposed to combine the advantage of ion engines (high Isp) with the advantage of chemical propulsion (high thrust), by mid-course refueling the chemically propelled, fast, manned ship by means of electrically propelled tankers that would be launched several years ahead of the manned mission (Figures 6 and 7).

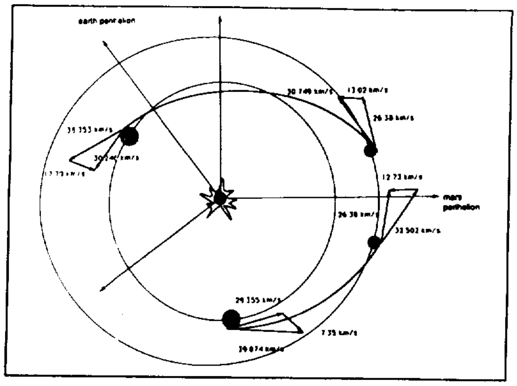


Fig. 6 Manned ship velocities in the Earth, Sun, Mars frames of reference

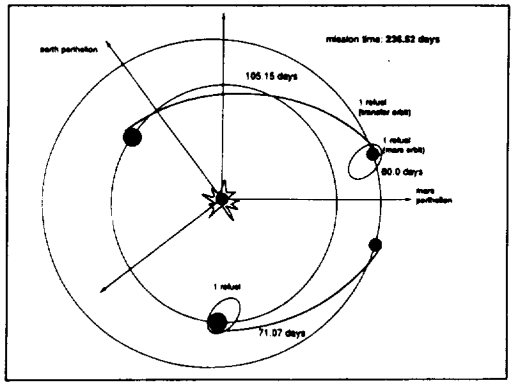


Fig. 7 Refueling points and elapsed days for the manned ship

Refueling a ship $n$ times is equivalent to an $(n + 1)$-fold increase of its Isp. Because of the very high Isp of the electrically propelled tankers, the total mass that must be assembled in LEO is decreased in comparison with more conventional mission scenarios.

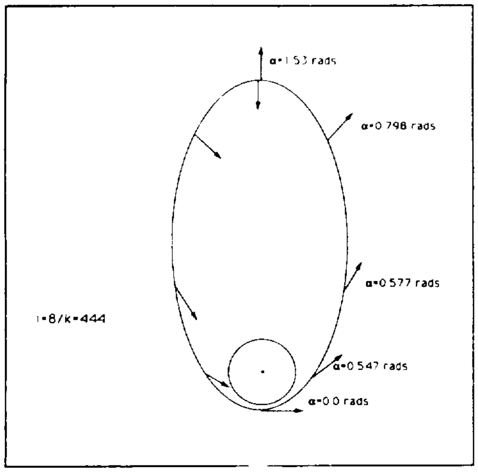


Fig. 8 Computed direction of the ion engine thrust needed for a sequence of orbits with constant periapsis radius

To allow rendezvous with a manned ship on a fast, hyperbolic trajectory leaving Earth or Mars, it is necessary for the tankers to follow trajectories which are characterized by a constant periapsis radius. This requires thrusting at an angle to the instantaneous direction of travel (Figure 8). Numerical studies of such trajectories have been carried out in sufficient detail to allow sizing of the ion engines and determining the propellant (argon) mass and the electric power requirement. A nuclear reactor of an upgraded SNAP-100 type was assumed, and the designs of power conversion equipment and radiators considered.

The difficult problem of long term storage in space of cryogenic propellants was considered, including the need for the re-condensation of the boil-off. The study also addressed the overall design of the tanker fleet, including their assembly in low earth-orbit.

Table 2 Mission events

| | |
|---|---|
| A. | Manned ship is fueled, leaves LEO, escapes Earth, begins Earth-Mars Transfer (EMT). |
| B. | Manned ship rendezvous with Tanker #2, boosts for EMT. |
| B/C. | Rendezvous with Tanker #3 during EMT. |
| D. | Manned ship aerobrakes and circularizes into Mars parking orbit. |
| E. | Manned ship descends to Martian surface; surface exploration. |
| F. | Manned ship rendezvous with Tanker #4, escapes Mars, begins Mars-Earth Transfer (MET). |
| G. | Manned ship rendezvous with Tanker #5, completes boost for MET. |
| H. | Manned ship retrofires with remaining fuel at Earth vicinity. |
| J. | Manned ship aerobrakes to capture at Earth and return to LEO. |

This project was divided into five areas of specialization:

**Trajectories:** Determine the most efficient paths to get the tankers to the proper place at the correct velocity, at the proper time. Start times, start positions, thrust directions, and coast times. **Power Systems:** Narrow down possible power system scenarios. Select equipment for the chosen scheme. Determine shielding needs if nuclear power is used. Optimize the design by minimizing mass. **Aerobraking:** Determine the feasibility of, and requirements for, aerobraking at Mars to position tankers 4 and 5. Find configuration design constraints. Find the aerobraking trajectories and the aeroshield temperature distribution. Analyze possible alternatives to aerobraking. **Thermal Control:** Consider energy management and thermal environment control. Cryogenic recondensation of boil-off. Analysis of heat transfer during different mission stages or events. **Mechanical Design:** Develop the general physical configuration of the spacecraft. Integration of subsystems.

## Design Specifications

Certain initial assumptions and estimates were made to allow concurrent trajectory and tanker design. Mass was estimated at 330 metric tons. A thrust level of 40 newtons was chosen. Ion engines capable of a specific impulse of 16,000 seconds were chosen for the main propulsion system. The mass of the manned ship is assumed to be 35 metric tons without fuel.

Since safety is a primary concern, the scenario proposes sending out more tankers than the minimum of five required for a first mission. The more tankers that are available en route to Mars, the greater the safety. A network of tankers would allow for aborting the mission at any time and would allow for possible mechanical malfunctions of a specific tanker during the refueling process. The scenario is modeled on the assumption that there will be subsequent missions using the same fuel and refueling process. The extra tankers would then be utilized in later missions.

The project focuses on the design of the tankers and their mission profiles.

The tankers are required to place 189 metric tons of LOX and 27 metric tons of $LH_2$ at the correct point in space, at the proper velocity, and at a specific time. In addition, the tankers must have the extra thrust capability to allow for a launch window of six days and a 6-day fueling opportunity for the manned ship. Auxiliary propulsion systems which allow for quick course changes must be provided for. The tanker must also bear the burden of maneuvering for docking.

Choosing the best source for the propellants is very important for this mission. These propellants will make up about 71% of the total tanker mass. The LOX alone will be 62% of the total tanker mass. The sources investigated were Earth, the moon, Mars, and Phobos. Sources were compared on the basis of the amount of mass which must be placed in LEO, development cost, initial equipment/mass investment, propellant transportation, and the probability of mission success.

Phobos may be the best source due to its location and extremely low gravity. The low gravity allows the tankers to take propellants directly from the surface rather than by rendezvous with chemically propelled surface-to-orbit transport vehicles. This method would be very time efficient for tankers 4 and 5 as only 30% of the total tanker mass must be transported to Mars vicinity. It may be possible to have tanker 5 bring the production and storage equipment to Phobos and first fill tanker 5 and then fill its own tanks. The Phobos facility would replace the fourth tanker, fuel tanker 5, and refuel the tankers heading back to earth.

Although Phobos appears to be the most efficient propellant source, in our baseline design oxygen produced on the moon and ground-produced hydrogen will be used. Later missions may exploit Phobos, but the added complications pose too great a risk to the initial missions and too much of an investment. In addition, LOX production on the moon is assumed to have already begun as part of a moon base project. The acquisition of LOX on the moon greatly reduces tanker mission time and surface-to-orbit transfer costs when compared with LOX brought up from the earth to LEO.

Tanker 1 will collect LOX at the moon and return to LEO. (The manned ship will be in LEO and will receive the $LH_2$ directly from earth.) Tankers 2, 3, 4, and 5 will receive $LH_2$ in LEO and then move out to the moon to receive the LOX. They will continue on their missions without returning to LEO.

## Power Systems

The propulsion estimates translate into an electrical power requirement of approximately 4 megawatts.

Nuclear electric and solar electric means of producing the electrical power for the tankers were studied. Estimates of solar array size showed the required area to be larger than 40,000 square meters or roughly eight football fields. Initial weight estimates for the nuclear electric power system gave 24 metric tons. Therefore, the solar array, support structure, and servicing systems had to weigh less than 24 metric tons to be competitive. Achieving the required structural stiffness for such a sizable array appeared to be very difficult. In addition, the array orientation requirements and size severely restricted the tanker configuration. Aerobraking would be impossible, as a structure this large and fragile could not be folded behind the aerobrake and would be subjected to the g-loads required. The solar array would not produce the public concern associated with nuclear systems, and would require much less development time. Nuclear electric power was chosen for the tanker ships.

**Reactor.** An extension of the SP-100 program would offer the most suitable nuclear power system based on power-to-weight ratios. The reactor would be a lithium-cooled "pin-type" reactor with advanced "PWC-11" cladding and structure configurations. Heat pipes transport the thermal energy to the power conversion equipment.

**Radiation Shield.** Shield size, weight and shape are determined by the size of the nuclear reactor and the vehicle configuration. The shield does not have to be man-rated, which greatly reduces its weight. It is only needed for the protection of the electronic equipment during the tanker's lifetime. Efficient shielding is accomplished by placing a circular shield on one end of the reactor and placing the rest of the tanker in the conical safe zone. Increasing the distance of electrical systems from the reactor reduces the required shield thickness.

**Power Conversion System.** A potassium Rankine cycle and a free piston Stirling engine were studied as possible candidates. The potassium Rankine system is more developed and is lighter than the Stirling engine for the required power levels. However, the Stirling engine is believed to have potentially greater efficiency. This translates into a lower thermal power requirement which reduces the reactor size, and therefore the shield mass. In addition, less waste heat must be radiated which greatly reduces the radiator size and mass. The free piston is the only moving part and there are no sliding seals. The piston works with a linear alternator. Research into Stirling engines at high power levels is currently underway and is expected to be mature by the mission time. An axial opposing cylinder configuration will further reduce vibrations.

**Radiator.** Heat is transported to the radiator and distributed by a series of heat pipes. The radiator is conical in shape to stay just within the reactor radiation safe zone. A reflector plate may be added at the end of a cylindrical or flat sided radiator to create the conical safe zone, without affecting the radiator heat transfer rate.

## Aerobraking

The given constraints on our design were that the tankers should withstand a maximum of 5 g deceleration. The change in velocity can be a maximum of 8.5 km/s when entering the Martian atmosphere. Aerobraking was considered as an option for slowing when approaching Mars. This operation performs the necessary aero-assisted capture and orbit transfer by utilizing its aerodynamic surface to produce drag and some lift. Important factors in an aerobrake capture system are the flight path angle, the ballistic coefficient, the closure angle, which is found to be 22° from various trade-off studies, and the lift-to-drag ratio.

For successful aerocapture, planetary features of the Martian atmosphere are an important consideration. Density and temperature can change dramatically due to seasonal and weather changes such as the very frequent dust storms on Mars. Estimates of the Martian atmospheric density are presently uncertain. Early missions will be necessary to develop confidence in analyzing and predicting the planetary characteristics. The surface terrain such as mountain ranges are an important factor as well.

The initial research into aerobraking focused on necessary size, shape, thermal, and flight characteristics. High L/D aerobrakes were initially considered but were rejected due to their large masses. Biconics seem to have extremely high point heating that pushes material limits. Raked spherical cones have reasonable mass and heating characteristics, but the low L/D ratio complicates control. Ballutes are much lighter than other aerobrakes, but they require a coolant load that negates any overall mass savings as compared to the raked spherical cone. The raked-spherical cone was therefore chosen as the best option.

Material selection and construction of the aerobrake become very significant, especially considering possible fatigue and thermal expansion. Stagnation point temperatures over 2400° Kelvin were found for some entries. Mass of the aerobrake and heat transfer to the cryogens raise serious questions about aerobraking.

Nuclear thermal propulsion retro-firing was analyzed as a possible alternative to aerobraking. This system would need a very large amount of propellant that would make it much heavier than the aerobrake configuration.

Another possibility uses the ion propulsion system for deceleration as well as acceleration. This increases the mission and operating time but the tanker mass savings would be approximately 10% when compared with the aerobrake configuration. A combination of ion propulsion and low energy trajectory to Mars moves a 308 metric ton tanker in approximately 580 days. For the tankers, ion engine retro-thrust is a very favorable alternative to aerobraking.

## Thermal Analysis

Thermal analysis is crucial to the successful design of the tankers. It encompasses every aspect of the tankers' main functions and requirements. The most important considerations are those that deal with the fuel tanks for the liquid propellants. The liquid hydrogen and oxygen have very low vaporization temperatures, approximately 20° Kelvin for $LH_2$ and 90° Kelvin for the LOX. Any heat added to the tanks may cause the propellants to warm and vaporize, which could be catastrophic to the mission. The first objective was to design a successful thermal blanket configuration for the LOX tank, first with radiation effects taken into account only, then including conduction. It was necessary to examine different materials in order to select the best possible configuration for a multi-layer insulation blanket. An available option was to use very optimistic values for the radiative properties of the materials. For example, absorptivity values were used in the range of 0.04 for the top layer of the blankets in order to greatly decrease the heat flux.

The next main problem for the thermal analysis was to see if refrigeration cycles were needed to keep the propellants from vaporizing, and if so, to design a successful configuration. An idea proposed was to use concentric cylindrical tanks for the $LH_2$ and LOX. This concept was not used, however, because it brought up many complications including fuel transfer, and extra weight. If two separate tanks were being designed, a refrigeration system for at least the liquid hydrogen tank becomes necessary. Possible suggestions included the use of Stirling engines and sorption pumps. One system that was studied was the Molecular Absorption Cryogenic Cooler proposed by a design team from the Jet Propulsion Laboratory, using the Joule-Thompson process and using waste heat from the power conversion cycle (Figure 9).

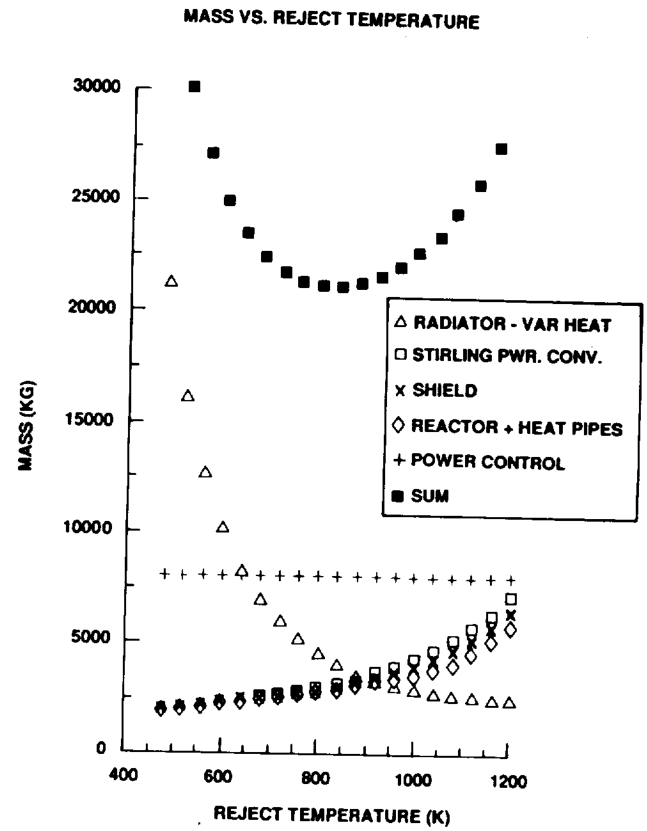


Fig. 9 Mass of refrigeration system vs. radiator temperature, for Stirling engine power conversion

Other thermal problems studied were the effects from excessive heating from the aerobrake, and the exchange of fuel. Thermal problems also could arise from the proximity of the tanks to the aeroshell.

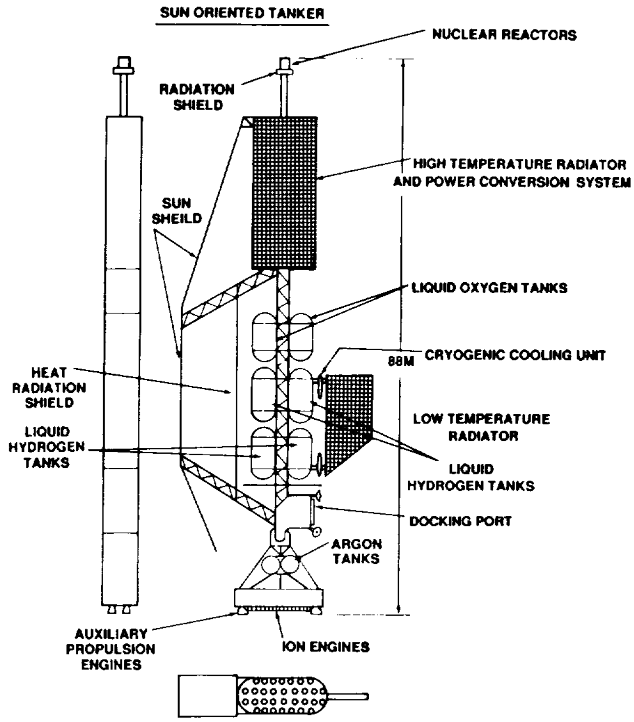


Fig 10 Sun-oriented tanker ship

## Mechanical Design

Emphasis was placed on the required mass in LEO as a rough indication of project cost. Other considerations included: (1) Maximize reliability, lifetime and reusability; (2) Minimize complexity; (3) Minimize the number of moving parts; (4) Minimize sliding seals; *no exterior sliding seals in gas storage systems*; (5) Minimize the use of flexible fluid lines; (6) Provide redundancy; (7) Connect independent systems in parallel; (8) Provide resistance for meteoroid damage; (9) Minimize on-orbit assembly; (10) Provide capability for emergency propulsion of the manned ship; (11) Provide docking clearance for ion and auxiliary engine exhaust cones; (12) Allow access for auxiliary engines to the main LOX and $LH_2$ tanks; (13) Place center of gravity of the docked configuration on a possible thrust vector; (14) Locate habited section of the manned ship in the radiation safe zone.

**Sun-oriented vs. axisymmetric slowly spinning ("Rotisserie Mode")**

A sun-oriented tanker (Figure 10) has the mass benefits of less tank insulation and of a smaller cryogenic cooling system. Thermal fatigue is minimal.

An axisymmetric tanker is more conventional and has more mission flexibility. The tanker's spin sets up forces that make the separation of liquid and vapor easier. The refrigeration system is larger than in the previous configuration and requires extra equipment to cool to ultra low temperatures. More tank insulation is also needed.

**Baseline design**

The baseline design (Figures 11 and 12) does not use aero-braking, is axisymmetric and thermally rolled about its axis of largest moment of inertia. The configuration is very stable and may rotate while docked. No configuration movements or adjustments are needed to accomplish all design objectives. Simplicity of control and a reduced $LH_2$ loss possibility were deemed more important than cooling system mass savings. The extra fuel and systems for the sun-oriented tanker tend to minimize possible mass savings. Microgravity vapor/fluid separation will make the refueling and refrigeration process more efficient.

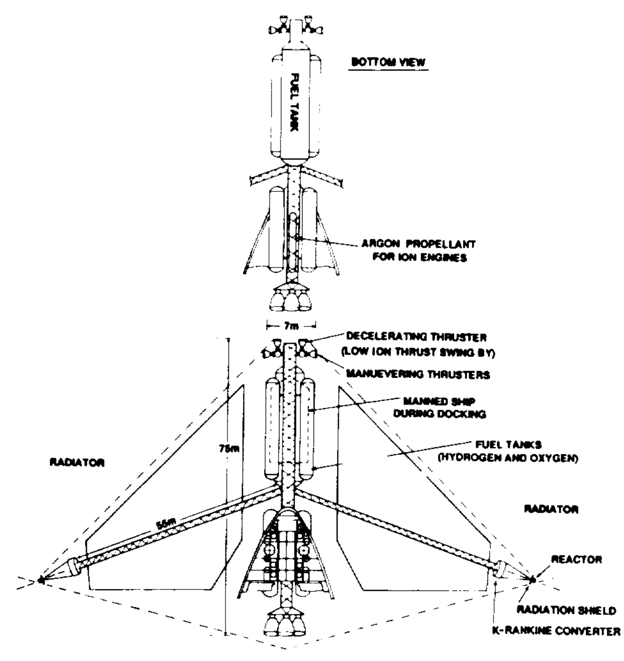


Fig. 11 Electrically propelled tanker, Alternative I

## Optimization

Once the general configuration was set, each subsystem was optimized for minimal mass. Figure 9 shows the optimization to find the best combinations. The individual curves were found by interpolating between estimates given in the literature and making certain assumptions. As an example, radiator mass was assumed proportional to area. The cryogenic storage system optimization followed the same procedure. Variables were tank wall thickness, $LH_2$ storage temperature (affects pressure), insulation thickness, low temperature radiator masses and sorbent pump mass. The power for the sorbent pump is reactor waste heat so the total power requirement is not affected.

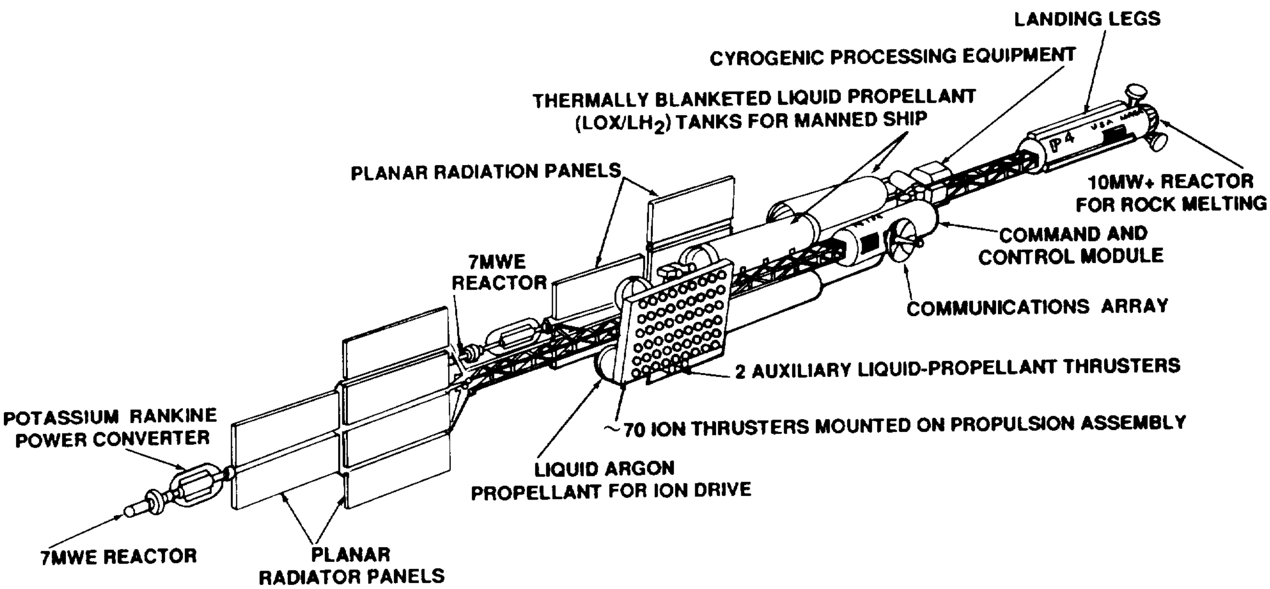


Fig 12 Electrically propelled tanker, Alternative II.

Table 3 Mass summary

| | |
|---|---|
| Power System | 27,000 kg |
| Propulsion<br>Includes auxiliary engines and control thrusters. | 12,000 kg |
| Cryogenic Storage System | 5,700 kg |
| Guidance, Navigation and Communications | 300 kg |
| Structure | 2,500 kg |
| Docking Unit/Miscellaneous | 3,000 kg |
| Argon | 35,000 kg |
| LOX (auxiliary propulsion fuel included) | 204,120 kg |
| $LH_2$ (auxiliary propulsion fuel included) | 29,160 kg |
| Fully loaded tanker | 318,780 kg |
| Mass in LEO (tanker) | 114,660 kg |
| Mass in LEO (mission) | 580,300 kg |

Table 4 Design summary

| | |
|---|---|
| Main propulsion | 60 electron bombardment ion engines. Specific impulse: 16,000 seconds |
| Power system | 4 MW .SP-100 type nuclear reactors (4). 1.4 MW free piston Stirling power converters (4) |
| Cooling system | Molecular absorption cryogenic coolers with precool systems |
| Truss structures | Ultra high modulus carbon fiber/epoxy tubes with aluminum end fittings |
| Tanker positioning is done mostly with the ion engines. *Aerobraking is not used.* | |
| LOX is acquired from the moon. $LH_2$ is brought from earth. | |