Preliminary Assessment of Artificial Gravity Impacts to Deep-Space Vehicle Design

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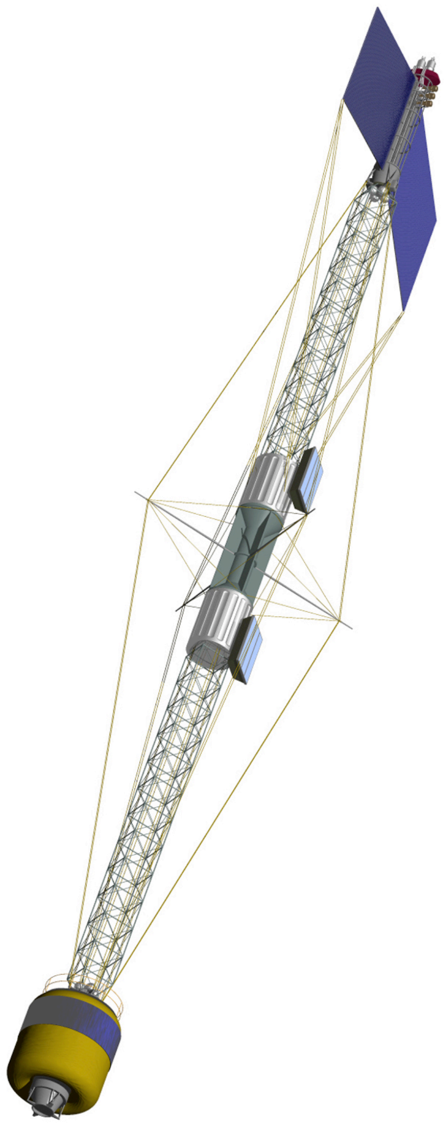

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Introduction

Even after more than thirty years of scientific investigation, serious concerns regarding human physiological effects of long-duration microgravity exposure remain. These include loss of bone mineral density, skeletal muscle atrophy, and orthostatic hypertension, among others. In particular, Ref. 1 states

> "…loss of bone density, which apparently occurs at a rate of 1% per month in micro-gravity, is relatively manageable on the short-duration missions of the space shuttle, but it becomes problematic on the ISS [International Space Station]… If this loss is not mitigated, interplanetary missions will be impossible."

While extensive investigations into potential countermeasures are planned on the ISS, the delay in attaining full crew complement and onboard facilities, and the potential for extending crews' tours of duty threaten the timely (< 20 years!) accumulation of sufficient data for countermeasures formulation. Indeed, there is no guarantee that even with the data, a practical or sufficiently robust set of countermeasures will be forthcoming.

Providing an artificial gravity (AG) environment by crew centrifugation aboard deep-space human exploration vehicles, long a staple technique of science fiction, has received surprisingly limited engineering assessment. This is most likely due to a number of factors: the lack of definitive design requirements, especially acceptable artificial gravity levels and rotation rates, the perception of high vehicle mass and performance penalties, the incompatibility of resulting vehicle configurations with space propulsion options (i.e., aerocapture), the perception of complications associated with de-spun components such as antennae and photovoltaic arrays, and the expectation of effective crew micro-gravity countermeasures. These perception and concerns may have been overstated, or may be acceptable alternatives to countermeasures of limited efficacy.

Objectives

This study was undertaken as an initial step to try to understand the implications of and potential solutions to incorporating artificial gravity in the design of human deep-space exploration vehicles. Of prime interest will be the mass penalties incurred by incorporating AG, along with any mission performance degradation.

Ground Rules

Artificial Gravity Parameters

In order to establish design requirements and constraints for an artificial gravity spacecraft, past ground-based and space-based research was reviewed. The parameters gravity-level (a_{AG}) and rotation rate (ω) are crucial to the feasibility of AG spacecraft designs, since they determine the required rotational radius: $a_{AG} = \omega^2 r$.

It must be noted that *there is essentially no data*, either through experiments or analyses to give any indication of the effectiveness of partial-gravity in ameliorating the physiological effects of microgravity. Additionally, such experimental data would be exceedingly difficult, time-consuming, and expensive to gather from human test subjects, requiring something like a space-based, variable-gravity rotating facility. For these reasons, *this study assumed an AG level of 1-g was required*.

Similarly, there is *no data* indicating that a crewmember could be centrifuged for a limited time (i.e., on a "daily" basis) in order to avoid deleterious microgravity effects. This data would be difficult to attain for the reasons stated above, and it is possible that the crewmembers would experience the stressful (and unpleasant) effect of readaptation at each cycle. Therefore, *this study assumed that the crew would be under nearly continuous centrifugation throughout the mission.*

The U.S. ground-based facilities that were capable of performing centrifugation research with human test subjects included the Pensacola Slowly Rotating Room (in operation from 1960-74) and the Rockwell Rotating Test Facility (1970). In the Soviet Union, the "MVK-1" and "Orbita" facilities served similar purposes. These facilities allowed test subjects to be centrifuged at varying rotational rates for weeks at a time, permitting assessment of motor skills, adaptation, and physiological effects. The main concerns involve the "cross-product" or "coriolis" accelerations experienced while an object is moving relative to the rotating environment. For humans, this manifests itself as accelerations sensed by the vestibular system (due to, for example, head movements) without corresponding visual cues, resulting in symptoms akin to motion sickness. The subjects experienced total (vector sum of induced and terrestrial) gravity levels of 1 to 1.4 g's. The results of these studies are summarized:

> "…at a speed of 4 rpm, some individuals will be naturally immune to motion sickness while others will have motion sickness but will adapt after a few days and suffer little decline in performance." (Ref 2)
>
> When rotation ranges from 3 to 6 rev/min … the initiation of rotation will elicit changes in postural equilibrium as well as symptoms of motion sickness, the extents of which are a function of the magnitude of the angular velocity. Nevertheless, adaptation can be achieved under these conditions in 6 to 8 days, and the remainder of the stay in the rotating environment is characterized by normal health and performance." (Ref. 3)
>
> "…ground-based results can be extrapolated to the spaceflight environment only when the AG in that environment is equivalent to 1 g." (Ref. 3)

Based on these conclusions, *this study baselined a maximum rotational rate of 4 rpm.* The impact of this assumption on spacecraft design practicality should be stressed. At 4 rpm, an AG level of 1-g is achieved with a rotational radius of 56 meters. If the acceptable rate were, for example, only 1 rpm, the required radius would be _ mile!

Finally, there were several space-based experiments that indicated the efficacy of artificial gravity. The Soviet Kosmos 782 (1975, 19 day fight) and Kosmos 936 (1977, 18 das flight) flew a facility in which ants, turtles, rats, plants, and cell and tissue cultures were centrifuged at 1-g, along with a 0-g control group. Postflight examination indicated the "artificial-gravity groups showed no evidence of typical adverse effects of microgravity" (Ref. 2). Also, in 1985 the Shuttle/Spacelab D-1 mission flew a biorack centrifuge containing seeds, bacteria, and human blood cells. These results were summarized: "microgravity effects at the cellular level may be eliminated by artificial gravity" (Ref. 2). *This study assumed that a centripetal acceleration of 1-g would be physiologically equivalent to a gravitational acceleration of 1-g (excluding coriolis effects).*

Mission Archetype

To evaluate a conceptual spacecraft design, some sort of mission parameters must be established to allow systems trades such as propulsion, power, habitation, etc. and establish the impact of an artificial gravity configuration. It was the intent of this study to retain a certain level of mission-independence, allowing the results to be applied to a range of destinations and mission classes. In reality, the combination of attainable propulsion technologies and potential destination distances which equate to flight times requiring artificial-gravity led naturally to round-trip Mars missions as a mission "archetype".

More specifically, this study adopted a Mars "opposition-class" mission, typified by an 18-24 month round trip with up to three months spent in the Mars system. This trajectory class can stress the interplanetary "steering" requirements, which may be a concern for rotating spacecraft. Also, these types of missions are challenging from the standpoint of propulsive performance, and it is desired to establish compatibility between AG and advanced propulsion technologies. In addition, a "split" mission approach was chosen, meaning that the crew transfer spacecraft does not bear he burden of transporting elements such as planetary landers, surface habitats, etc., which are assumed to be delivered by separate means, presumably on lower-energy trajectories. This allows some freedom in the spacecraft configuration, avoiding constraints imposed by less defined mission goals.

Previous design studies treated artificial gravity as a design requirement that was often dependent upon other parameters, specifically, propulsion technologies. Often times, an AG option was "tacked on" to propulsion choices made *a priori*, with questionable compatibility. In this study, AG was considered the driving requirement, with other system choices made (within "technology horizon" constraints) to be most compatible. One of these was nuclear electric propulsion (NEP).

NEP performance is characterized by relatively low-thrust, but high efficiency. This low thrust level should allow vehicle thrusting while under rotation due to the resulting small forces and torques, obviating the spindown-burn-spinup sequences required by high thrust systems (however, techniques for continuous thrust vectoring must be established). There may be inherent vehicle configuration synergies between NEP and AG. Typically, NEP vehicle designs require long masts or trusses to separate the nuclear power source from the regions of crew habitation (this "$1/r^2$" radiation shielding can be very mass-efficient, given light-weight masts). Such structures may also serve as the AG rotation "arms". Finally, as described below, the mass of the power production and conversion systems may serve as a good "counterweight" for the crew habitation systems, allowing a highly synergistic vehicle configuration.

To avoid conclusions regarding AG feasibility being influenced by questionably optimistic propulsion technology assumptions, this study established a "technology horizon" or initial operating capability of ~2015. This helped establish some of the key NEP performance parameters, enabling initial vehicle configuration concepts.

In Mars mission studies, the departure and return orbits at Earth are typically chosen to reflect the capabilities of the selected propulsion system. The arrival/departure orbit at Mars is usually chosen to reflect trades between the propulsive characteristics of the transfer vehicle and the lander. NEP vehicles typically exhibit poor performance deep in planetary gravitational fields since the low thrust levels translate into higher gravity losses and long orbit transfer times. For this reason, these studies assumed a high departure and return orbit at earth, specifically, the Earth-Moon "L_1" Lagrange point. This location may be synergistic with other human exploration goals, and as nuclear systems provide the performance capability for a reusable transfer vehicle, this staging location may be compatible with the operational characteristics of reusable space nuclear systems. Trades involving the assembly and delivery of the transfer vehicle to L_1 are not addressed in this preliminary analysis. The Mars orbit selection has been left open to trades in this study, but it was not evaluated in detail. Trajectory design of optimal low-thrust insertion into planetary orbits is a complex analysis, and will be addressed in future tasks. This study approximated the time and propellant required for transfer vehicle descent to and ascent from various circular Mars orbits.

Finally, this study assumes that a sustainable Mars exploration program is desired. As the vehicle under consideration will represent a considerable investment, and because nuclear systems have inherently high energy content, a vehicle reusability requirement of greater than three round trip missions is assumed.

Previous Studies

As stated, the number of past vehicle engineering studies designed to incorporate AG is not large. Two, however, were deemed to have requirements similar to those outlined above, and were examined for configuration concepts and operational strategies. The main differences in the two concepts centered

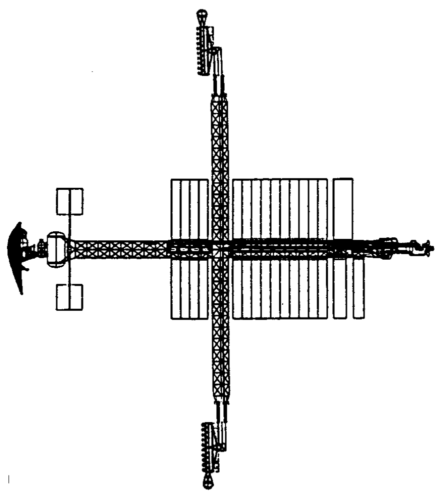


Figure 1. Ref. 4 Vehicle Configuration

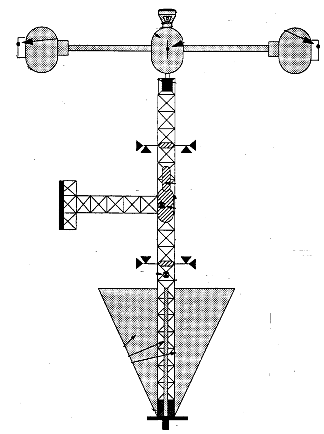


Figure 2. Ref. 5 Vehicle Configuration

on the system masses used to counterweight the habitation volume during rotation. Ref. 4 (Fig. 1) utilized the mass of the nuclear power generation and conditioning systems, while Ref. 5 (Fig. 2) split the habitation volume. Both concepts feature despun propulsion systems in order to allow thrust vectoring without requiring the precession of the angular momentum associated with the rotating sections. The strategy was to align the rotation plane with the interplanetary trajectory plane, as most optimal low-thrust profiles produce planar trajectories. While this may alleviate one design issue, another presents itself. Large mechanical rotation joints are required with continuous 100 kilowatt- to megawatt-level power transmission across the interfaces. While such mechanisms are undoubtedly technically feasible, the mass, complexity and reliability of such devices may prove challenging.

Approach

This study opted to initially focus on a simpler configuration which would potentially eliminate the need for large rotating interfaces, and examine the dynamics issues involving precession of the entire rotating vehicle for thrust vector control. To accomplish this efficiently, three top-level design goals need to be met: 1) utilize the power production and conditioning systems as a counterweight to the habitation volume to avoid ballasting or inefficient splitting of the habitat, 2) operate the power systems at gravity levels of ~1-g to simplify system qualification, and 3) achieve the propulsive performance necessary to accomplish the archetype mission with technology assumptions consistent with the "technology horizon". The implications of these goals are: 1) the power system mass must be nearly equal to the habitation system mass, and 2) the power system can assume a specific power level (α) of 4-8 kg/kWe and the propulsion system a specific impulse (Isp) of 4000-6000 sec.

Based on past NEP mission analysis data (Ref. 6,7,8), and habitation module design studies (Ref. 9) it appears that all of these design goals can be met.

Figure 3 illustrates the relationships among the parameters. The resulting vehicle power levels will lie in the range of 4-8 MWe.

The initial vehicle design is illustrated in Figure 4. The design trades that led to this configuration will be discussed in the next section.

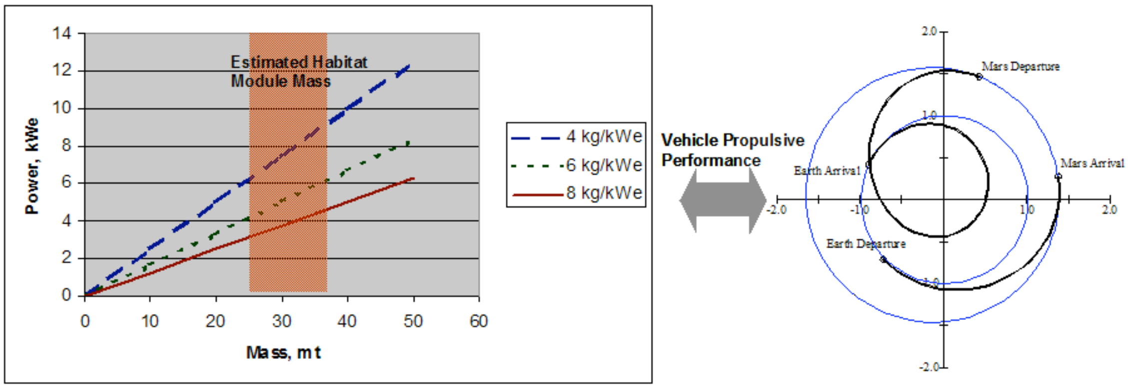


Figure 3. Design Parameter Relationships

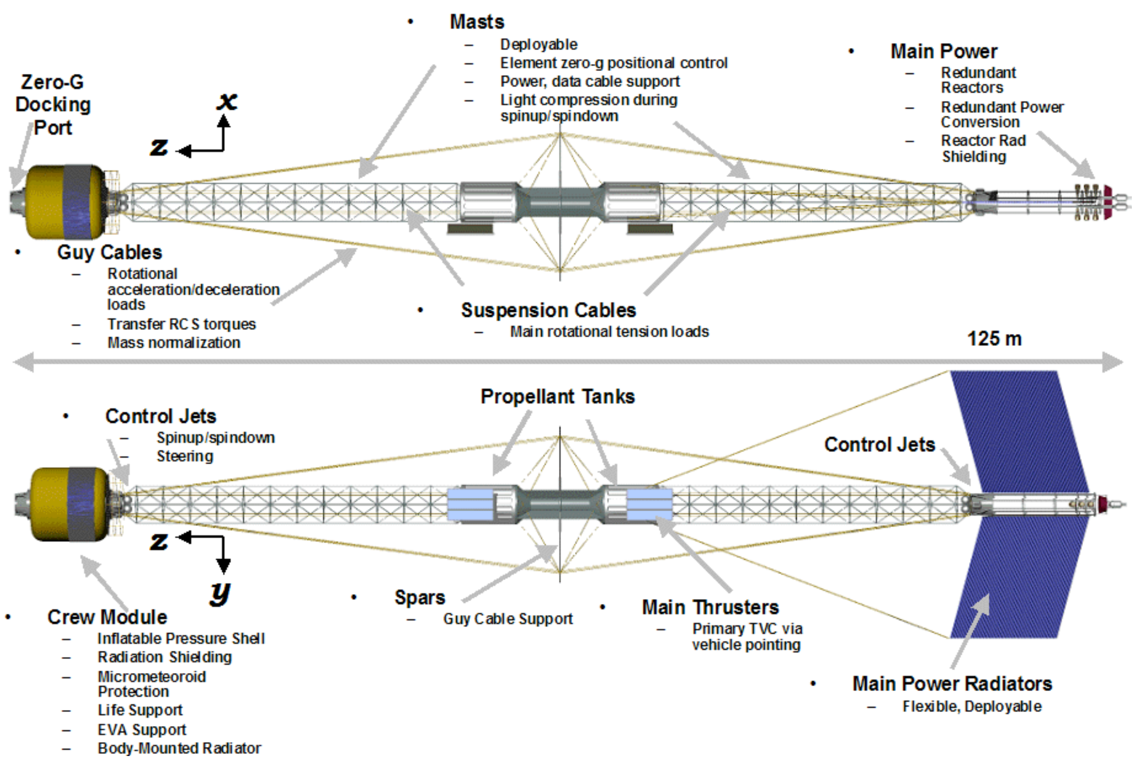


Figure 4. Initial Vehicle Design Concept

Study Results to Date

Trajectory Analysis

Because the trajectory class specified in the mission archetype displays significant variation in energy requirements over the Martian synodic period, a vehicle performance assessment was made for a representative "good" opportunity (2018) and "poor" opportunity (2026). Because low-thrust trajectory optimization is still a somewhat labor-intensive process (Isp, power level, specific power, and flight time can all be independent variables in the optimization process), three separate groups with three different analysis tools supported this activity. A group at the Johnson Space Center utilized a tool call RAPTOR, which is based on calculus of variations with a genetic algorithm to find reasonable initial control functions, the Glenn Research Center used VARITOP, also using a calculus of variations approach, and Science Application International Corp. brought CHEBYTOP to bear, a parameter optimization program based on Chebyshev polynomial approximations to the control histories. The results we compared to understand both the trajectory characteristics and any biases introduced by the individual tools.

These analyses indicated that the archetype mission can be accomplished within the power, specific impulse, and specific power ranges desired for the vehicle systems. Example mission performance results are shown in Figure 5. In each case, the stay-time at Mars was constrained to be no less than 90 days. The overall mission flight time in the "poor" opportunity was at the upper end of the desired goals. Shorter flight time may be achievable by increasing the vehicle power level, but this would imply a more technically challenging α to maintain the desired habitat counterweight. Alternatively, there may be trajectory techniques, including additional thrust arcs and Venus gravity assists on the return legs, which could increase performance.

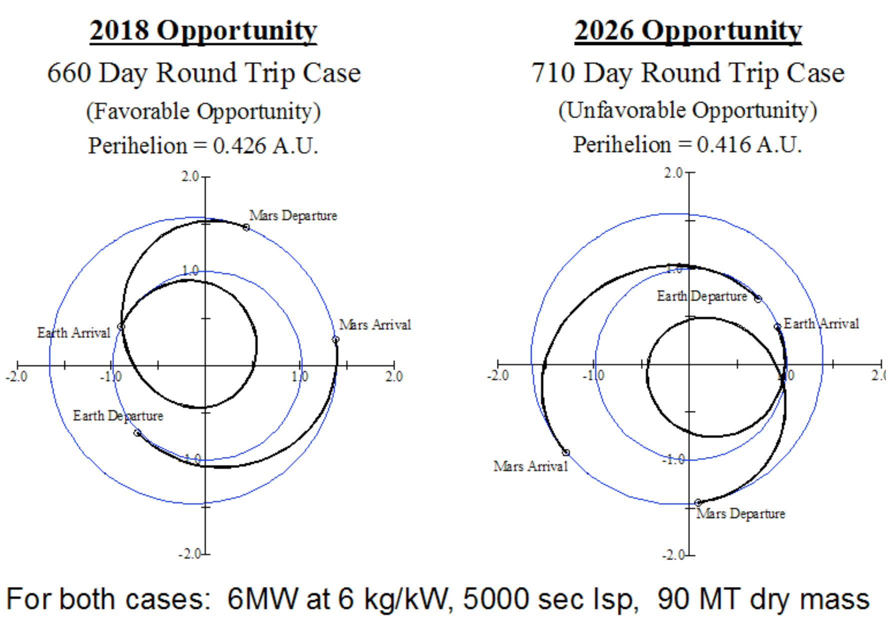


Figure 5. Representative Mission Performance

The return legs of these trajectories typically result in ~0.5 A.U. perihelia. While this may sound somewhat daunting, analysis has shown the thermal control capabilities of both the habitat and power conversion systems to be acceptable. These conditions may also be somewhat alleviated by the trajectory design technique mentioned above.

Dynamics

The inherent stability of objects rotating about particular axes is determined by the ratio of the object's principle moments of inertia as illustrated in Figure 6. The vehicle concept shown in Figure 4 is obviously a "major axis spinner", although the near symmetry about the z-axis may result in some level of active "roll" control requirement. This symmetry, combined with the location of the propellant tanks near the axis of rotation, should minimize the vehicle's angular momentum to the degree possible, allowing maximum maneuverability while under rotation, and minimum spinup/spindown effort.

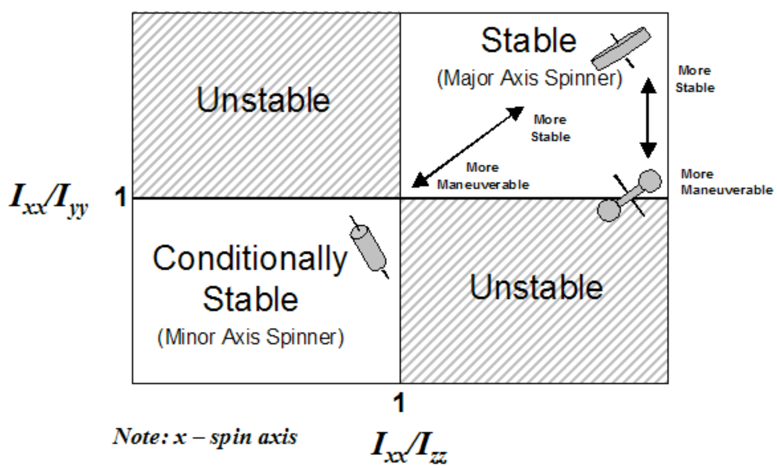


Figure 6. Rotational Stability

The vehicle spinup/spindown requirements are not particularly difficult to meet. Note from Figure 4 that the control jets are located such that they possess considerable moment arms. One trade that can be made is between spin thrust level and thruster on-time. If extended spinup times are acceptable, electric arcjets may have a role to play in this function. A thrust level of ~10 N would be adequate to establish a 4 rpm rotation rate in around two days, utilizing 100 kWe arcjets (assuming 30% jet efficiency). Abundant onboard power should be available since the main vehicle thrusters would probably not be utilized during spinup. The advantage of arcjets would be propellant reduction as illustrated in Table 1.

The primary parameter that will determine the feasibility of the vehicle configuration under consideration is the steering requirements during the mission. Recall, that to eliminate despun vehicle components and mechanical rotational interfaces, it was proposed to precess the angular momentum of the entire

Table 1. Vehicle Spinup Propellant Requirements

Thruster Isp, sec	Prop mass for spinup (or down), kg
310 (MMH/N2O4)	580
450 (LOX/LH2)	400
800 (Arcjet)	222
1000 (Advanced Arcjet)	180

Total moment = 2*Thrust*Moment arm
Moment arm = 50 m
Vehicle I_{xx} =2.1x10^8 kg-m^2

spacecraft in order to adjust the thrust vector. The trajectory analysis indicates that the steering requirements seem to fall into two classes – very slow rates (<2°/day) during the majority of the heliocentric trajectory, and moderate rates (10-15°/day) during Earth departure and arrival and during midcourse thrust reversals. This dichotomy suggests that different steering strategies may be pursued for these different mission phases. Higher rates would not be anticipated unless descent to lower Earth or Mars orbit was required.

This precessional steering would be accomplished by torquing the rotating vehicle at right angles to the desired steering direction. This torque would need to be applied intermittently during the proper phase of the vehicle rotation. Two different techniques could be utilized: 1) firing control thrusters, or 2) differentially throttling the main propulsion thrusters, as illustrated in Figure 7. The effectiveness of each of the methods will be examined.

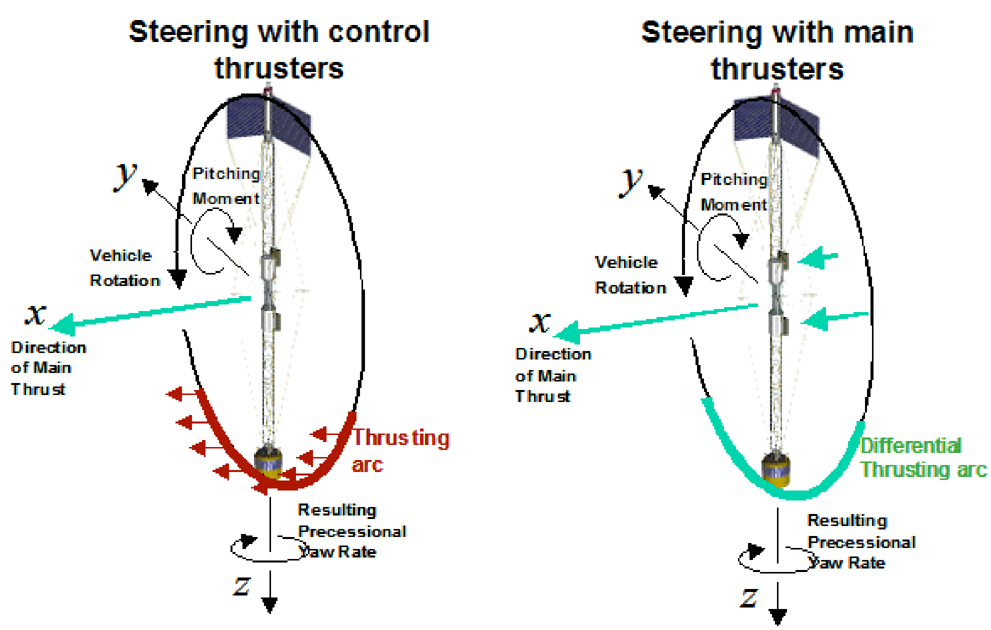


Figure 7. Precessional Steering Options

The effectiveness of control jet steering can be estimated by integrating the precession equation and substituting the control jet efficiency parameters. This indicates (not unexpectedly) that propellant quantity requirements can be relatively high, especially for chemical systems as shown in Table 2. In fact, if the steering for the entire mission was accomplished in this manner, the total requirement could exceed 15 tons (assuming 1440° of total turning). On the positive side, if the jet firings are implemented as non-coupled and always in the direction of flight, the thrust not only torques the vehicle, but also adds to its overall ΔV. This effect is shown in the last column of Table 2.

Table 2. Control Jet Steering Propellant Requirements

RCS Isp, sec	Prop. for 360° yaw, kg	Normalized for main prop. savings, kg.
310	4000	3690
450	2760	2450
800	1550	1240
1000	1240	930

$$\Delta\psi = \frac{g I_{sp} m_{prop} r}{I_{xx}\omega_S}$$

The thrust level required is a function of the required turning rate. For the moderate rates (10-15°/day), 10-15 N of thrust is required if a pulse is applied every 180° of vehicle rotation. For low rates, only 2-3 N is required. Again, arcjets may be applicable for this function, as the thrust levels, power requirements, and duty cycles are reasonable for this propulsive technology. Figure 8 shows the relationship of thrust, power requirements, and resulting turn rates.

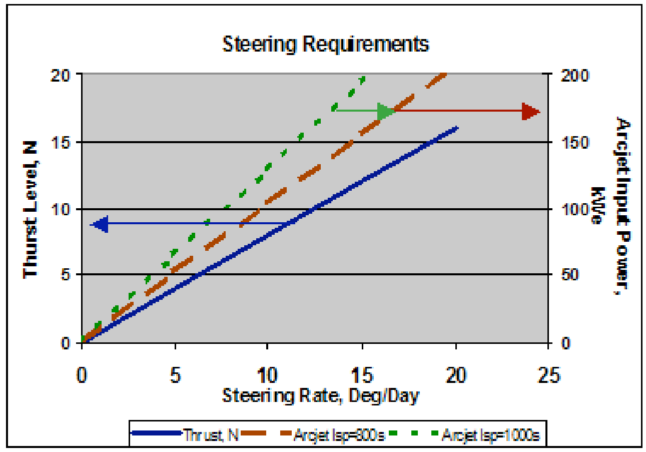
The second steering technique uses moments generated by differentially "throttling" the main electric propulsion thrusters during powered flight. This can be accomplished by either varying the propellant flow rate to the thrusters at a constant power input, or by varying the thruster power input at a constant flow rate. Additional main propulsion analysis will be required to make a definitive selection, but in this study, the former technique was assumed. In either case, it should be kept in mind that *steering by this technique uses essentially no additional propellant.*

The steering effectiveness of this method and the amount of throttling required will be a function of thruster location on the vehicle. The farther from the spin axis they are placed, the greater the turning effectiveness. However this would result in long feed lines from centrally located propellant tanks (recall the propellant was located near the spin axis to reduces the vehicle's moment of inertia). For this study, the thrusters were located near the tanks, with thrust offset 10 m from the vehicle spin axis. To attain the low, interplanetary steering rates (~2°/day), a ±5% thrust variation every 180° of vehicle spin was required. This equates to a thrust level variation of ±5 N per thruster produced by a propellant flow rate variation of ±0.25 grams/sec. Figure 9 shows results of a numerical simulation of this steering technique.

The selected vehicle configuration makes one additional steering technique possible. If a nearly 180° steering change is required, the vehicle could be rotated about its minor axis (z-axis in Figure 4). This could provide a relatively rapid reverse in thrust direction, without slewing the vehicle's angular momentum. Another possible implementation of this technique could be a second set of main thrusters with a "-x" thrust direction, eliminating the need for the minor axis rotation. The applications for such a maneuver would be the mid-course "turnarounds" and limited planetary "spiraling".

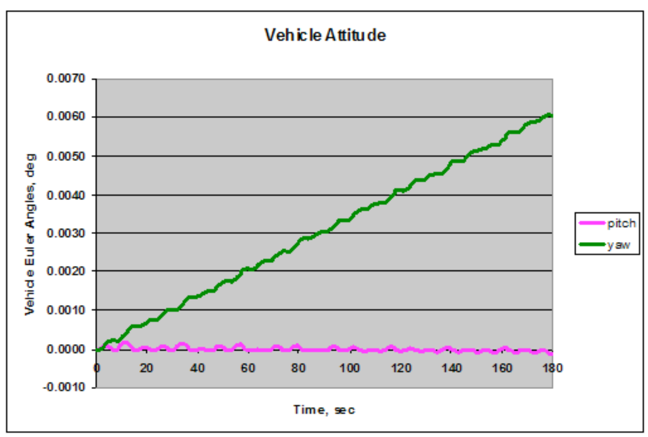
To formulate an example steering strategy, the mission profile was divided into segments where the three different steering techniques described above could be used to their greatest advantage. Table 3 shows that by utilizing control arcjet "impulse" steering for the moderate rate maneuvers, main thruster steering for the low rate maneuvers, and minor axis rotation for the 180° maneuvers, the steering propellant requirements can be reduced from the initial estimate of 15 tons to around 1 ton.

$$\dot{\psi} = f\,\frac{r\,T_a}{I_{xx}\omega_S}$$



Moment arm = 50 m
Pulse applied every 180°
Pulse "Width" = 90° of arc (90% thrusting efficiency)
Arcjet Eff. = 30%
Vehicle I_{xx} =2.1x10⁸ kg-m²

Figure 8. Control Jet Steering



Moment arm = 10 m
Throttle "doublet" applied every 180°
Pulse "Width" = 90° of arc (90% thrusting efficiency)
EP Thruster Eff. = 60%, Nominal Isp = 4000 s
Constant EP Power = 6 MWe
Vehicle I_{xx} =2.1x10⁸ kg-m²

Figure 9. Main Propulsion Steering

Table 3. Steering Strategy

Mission Phase	Maximum Turn Required	Maximum Required Turning Rate	Impulse Steering Only (ArcJet)	Impulse + Minor Axis Rotation	Impulse + MAR + Main Propulsion Modulation
Earth-Moon L_1 Departure	180°	15°/day	620 kg	620 kg	537 kg
Heliocentric Outbound, 1st arc	65°	2°/day	224 kg	224 kg	0
Mid-Course Thrust Reversal	180°	~10°/day	620 kg	TBD (small)	TBD (small)
Heliocentric Outbound, 2st arc	65°	2°/day	224 kg	224 kg	0
Mars-Sun L_1 Arrival	small	small	~0	~0	~0
Spiral to/from High Mars Orbit	Multiple revs	288°/day slew (Deimos) 180°/hr MAR	Impractical	TBD (small)	TBD (small)
Mars-Sun L_1 Departure	180°	2°/day	620 kg	TBD (small)	~0
Heliocentric Inbound, 1st arc	225°	2°/day	775 kg	775 kg	0
Mid-Course Thrust Reversal	180°	~10°/day	620 kg	TBD (small)	TBD (small)
Heliocentric Inbound, 2st arc	225°	2°/day	775 kg	775 kg	0
Earth-Moon L_1 Arrival	180°	15°/day	620 kg	620 kg	537 kg
			5098 kg	**3238 kg**	**1074 kg**

Structures

It is evident that spacecraft extended structures of some type will be necessary for the 1-g, 4 rpm AG operation. These structures must be lightweight to maintain propulsive performance, must be somewhat stiff and strong to support the centripetal tension loads and to transfer propulsion forces and moments, and must be deployable or extendable for practical assembly scenarios.

Initially, a "suspension-compression" structure was proposed using cables for counterweight mass support during spin, guy cables and spars for moment transfer from the outboard control jets, and an erectable mast for positional control of vehicle modules (no spin) and compression loading during the initial stages of spinup and final stages of spindown. The material selected for the cabling was liquid crystal polymer fibers due to their large specific tensile strength (16x steel) and their high resistance to abrasion, fatigue and radiation. For the masts and spars, ultra-high modulus graphite was selected for its extreme stiffness, large compressive strength and negligible thermal expansion. This was the concept shown in Figure 4.

The design for the main masts became somewhat problematic. These structures will only be transiently loaded in compression (on the order of 20 N at initiation of spinup). For AG operations, it serves no structural purpose, and matching the strain of the suspension cables with the zero-load mast length may result in complex positional mechanisms. A deployable, articulated mast also would not be appropriate for tension loading of the magnitude required by the AG vehicle if it were to replace the suspension cables, as the joints connecting the segmented longerons and diagonals would be prohibitively large and massive.

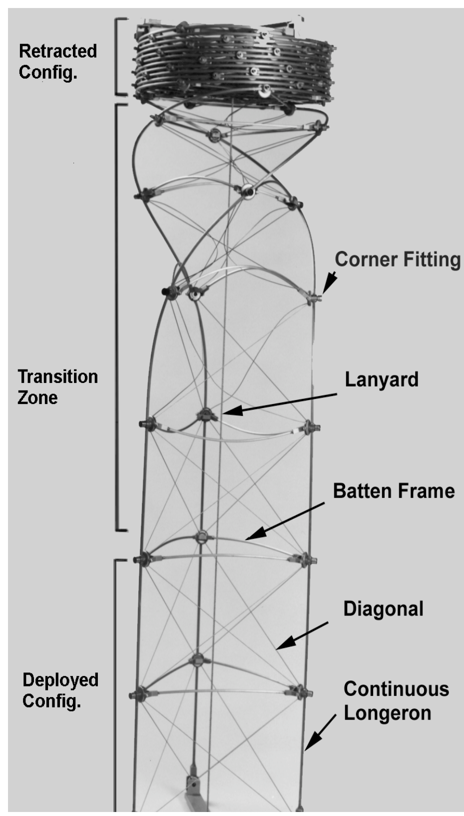


Figure 10. Coilable Mast Concept

A different approach was investigated. A "coilable" mast design using continuous pultruded uniaxial composite longerons is proposed. Such a design resembles a "rope ladder/tether" type structure in that it is not sized based on buckling strength, but rather by axial load capability. An important distinction is that such a structure can also resist bending and shearing loads. The graphite-epoxy fibers would be continuous along the length of the longeron and oriented optimally for axial stiffness. There are no joints along the mast to induce compression/tension hysteresis or failure modes, and there is essentially no "non-structural" material. An example of such a structure is shown in Figure 10.

The study contracted with Able Engineering to design such a structure which could support the tension loading of the vehicle AG operation, and could also transmit the forces and moment associated with the main propulsion system and the steering strategies identified above. An extremely lightweight (150 kg), compact (<1 m stowed length for a 50 m mast) design resulted. To reduce the strain energy of the packaged boom, the design uses a bundle of small-diameter rods instead of a single large rod for each longeron. This also provides structural redundancy and reduces the mast deployment push forces. This intrinsic push force is sufficient for deployment, with a motorized lanyard to pay out the masts (Figures 11 and 12). The Able report is included as an attachment.

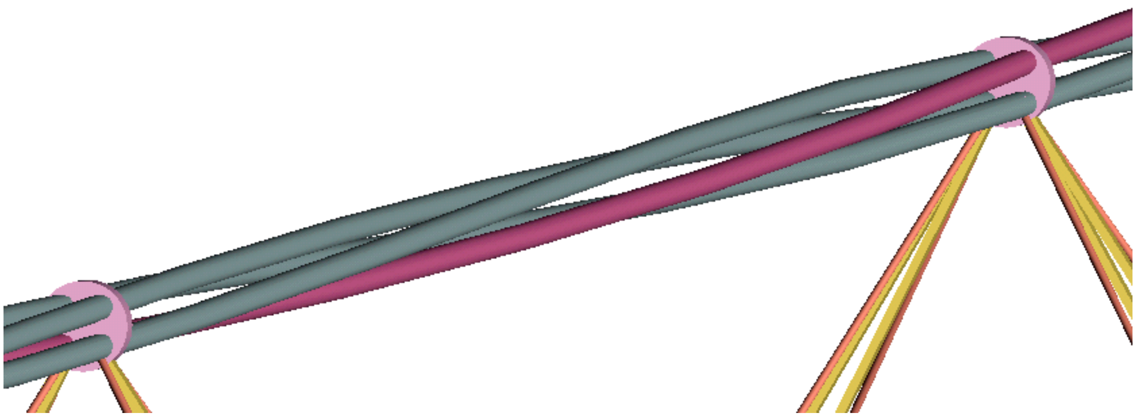


Figure 11. Longeron Bundle

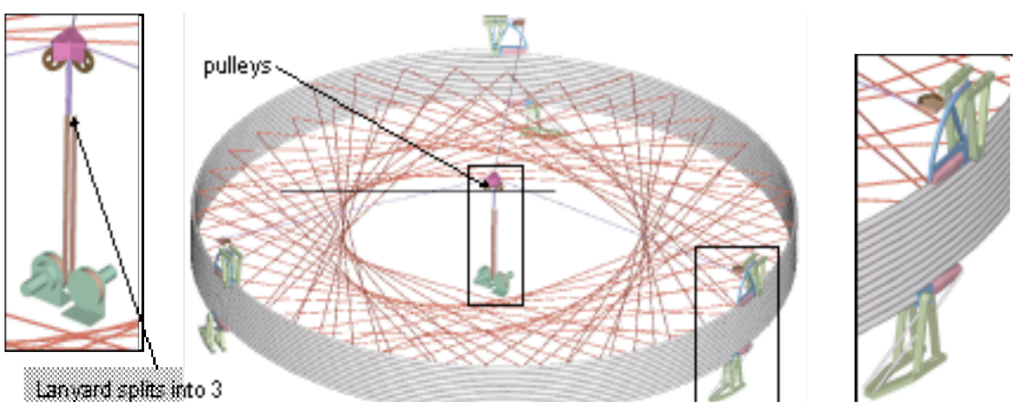


Figure 12. Stowed Mast and Deployment Mechanisms

Power and Propulsion

Three power and propulsion point scenarios were examined to understand the effects of reactor power, power conversion, and propulsion efficiency on the wet mass of the AG transfer vehicle. In addition, reductions in habitat and vehicle structural masses were assessed. All of the scenarios were able to accomplish the archetype mission. The results are shown in Figure 13. It should be noted that modest changes in these parameters can have the effect of halving the vehicle wet mass. For this study, the most conservative scenario (Scenario 1) was assumed, but this sensitivity indicates that future work should carefully examine the expected level of performance of these systems.

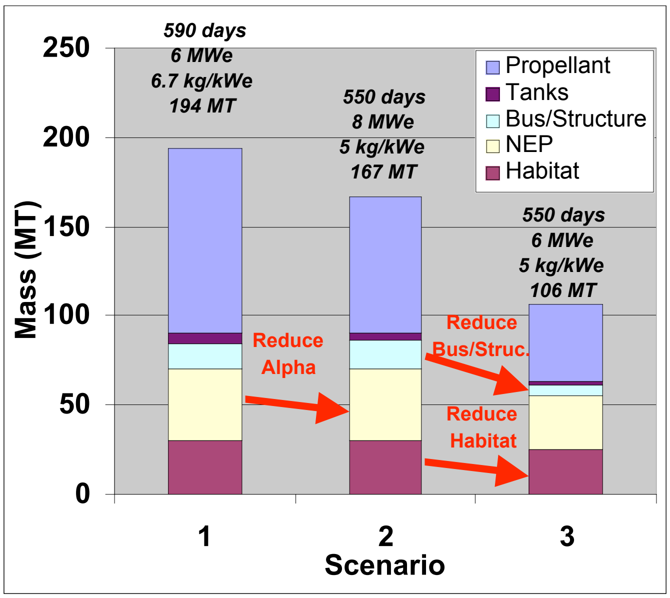


Figure 13. Power/Propulsion Scenarios

The reactor design used for assessment purposes was a 15 megawatt-thermal fast spectrum, boiling potassium reactor with a ceramic/metal core composed of uranium nitride in a tungsten/rhenium matrix (UN/W-25Re). The power system would utilize two such reactors, having a four-year life at full power operation. A potassium-Rankine power conversion system was chosen over other cycles, as this would result in lowest power conversion system mass at these power levels, the smallest radiators, and the lowest required reactor temperature. It was felt that these considerations outweighed the complexity of two-phase fluid management and liquid metal working fluids. The primary radiator would be 500 to 700 m^2 in area (assuming a rejection temperature of 1000K), and would be composted of carbon-carbon composite heat pipes with metal liners and potassium working fluid. A tungsten/lithium hydride reactor shadow shield is used to reduce the radiation exposure to less than 1 rem/year at 100m.

For system redundancy and possible compatibility with smaller power generation systems, the conversion system utilizes six one MWe turboalternators, each running from a separate fluid loop from one of the two reactors. The power output from the turboalternators would feed into a cross-strapped power management and distribution system and would subsequently power the electric thrusters. This system architecture provides for graceful degradation in the event of reactor, fluid loop, or turboalternator failures. The power system is illustrated in Figure 14.

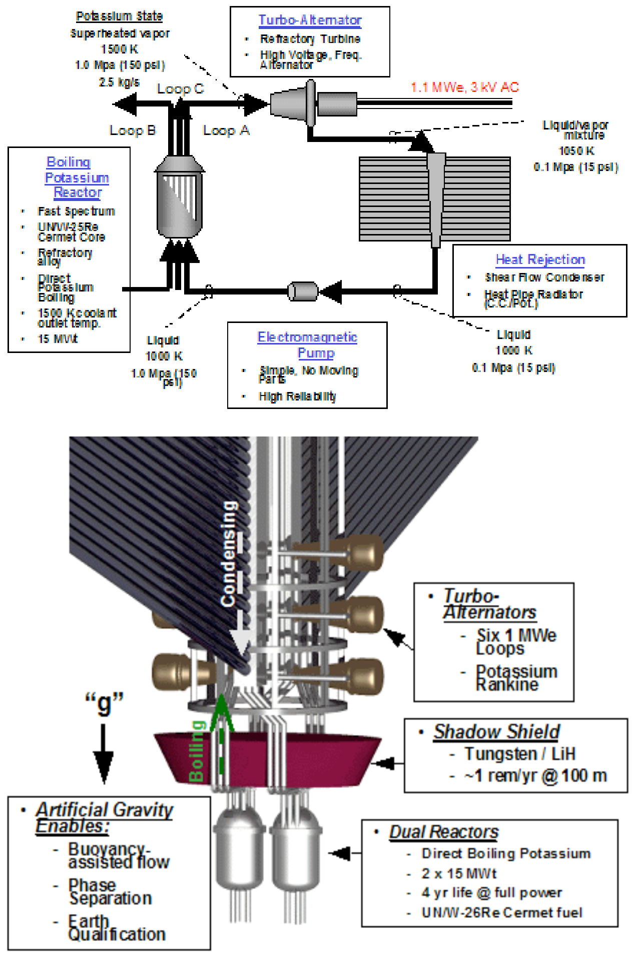


Figure 14. Power System

Three electric propulsion technologies were considered in this study: ion thrusters, magnetoplasmadynamic (MPD) thrusters, and RF induction plasma thrusters (VASIMR). For the fidelity of the current analysis, all of these systems have roughly the same performance and thruster efficiencies. Figure 15 shows the characteristics of a 1 MWe electric thruster. For this study, 60% jet efficiency was assumed.

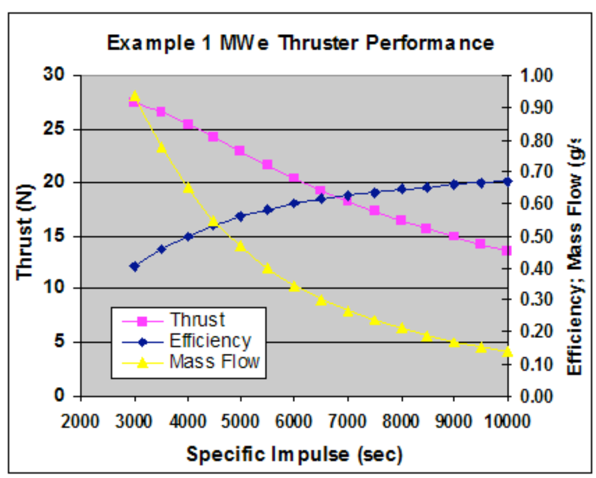


Figure 15. Thruster Performance

A more important characteristic may be the type of propellant used. Ion and MPD thrusters tend to use high-density propellants. This allows efficient propellant tankage and packaging near the vehicle spin axis. The propellant tanks in Figure 4 are sized for MPD thrusters (lithium, 500 kg/m^3) and would be even smaller for ion thrusters (argon, 1400 kg/m^3). The propellant of choice for VASIMR, however, is hydrogen, which would have severe configuration impacts for an AG vehicle. It may be possible to fuel a VASIMR thruster with denser fluids, such as deuterium or nitrogen and alleviate some of these issues. VASIMR thrusters were not examined in this study.

Habitation

As mentioned earlier, habitation module designs for missions of the type being considered in this investigation are available from past studies (Ref. 9). Two major differences justified a reexamination, however. One, of course, is the 1-g operational mode. The other is the availability of abundant power from the spacecraft's nuclear power system. Typical power requirements for habitats run in the 10's of kWe - less that one per cent of the reactor power output. For each of the major habitation subsystems, the effects of these two environmental conditions were evaluated. The full report of the habitat module design is included as an attachment.

The following additional architectural level assumptions were made in order to provide necessary guidelines for system leads to develop their concepts:

- Time duration per mission: 18 months
- The habitat will support 6 people
- The initial operational capability will be between 2015-2020
- The transfer vehicle will be reused for subsequent missions
- The vehicle will not be required to perform any aerobraking or entry maneuvers
- Outfitting missions are acceptable
- EVA will be a required function
- There will be no re-supply of consumables during the 18 month mission
- The launching configuration of the habitat portion of the spacecraft should be no larger than 5m X 15m.

Structures

The structure and shell are to provide a safe habitat for the crew and the necessary space to store supplies and equipment to sustain them for the duration of the entire mission. The inflatable module design was chosen because it is the best means to effectively increase the habitable volume of a spacecraft while keeping the diameter of the core within acceptable payload size limits. The airlock system is to provide the crew with the capability to perform extravehicular activities. It is located atop the habitat module, so as to allow the fully suited EVA crewmembers easy egress from the module without climbing stairs, ladders, etc.

The primary impact of artificial gravity is the necessity to modify the core into a load-bearing structure. Previous inflatable module concepts had a structure suitable for launch, and were then reconfigured significantly to operate only in microgravity. They contained cloth flooring with inflatable supports, which would be insufficient in a 1-g environment. One solution would be strong, but lightweight composite isogrid deck panels supported by cables (Figure 16). The inner wall of the shell itself should remain unaffected by the 1-g acceleration, however the outer layers may sag outward, thereby compressing the outer shell and reducing the amount of MMOD and radiation protection towards the top of the module.

Thermal Control System

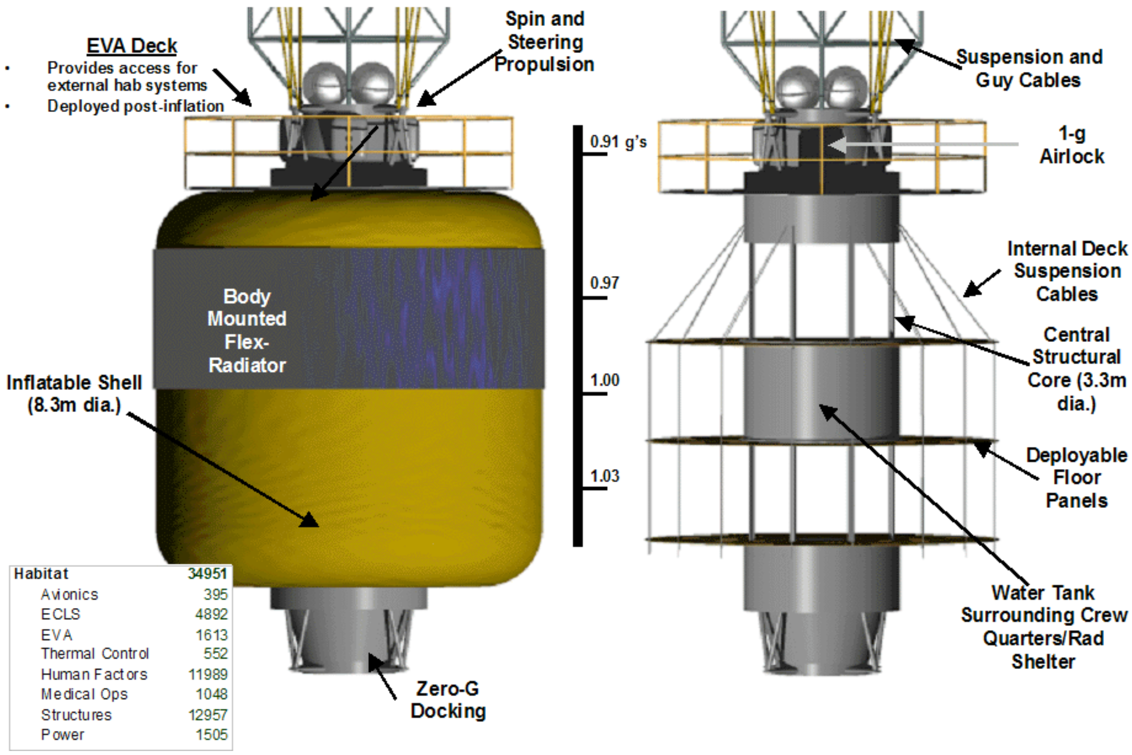


Figure 16. Hab Module Structural Concept

The TCS system concept makes use of flexible lightweight body-mounted radiators that are attached to the outer surface of the inflatable module. The TCS has been sized to collect and reject 15.0 kW of heat. A propylene glycol/water coolant is circulated inside the module to collect heat from heat exchangers and coldplates and this heat is rejected to space through the radiators.

A key issue is the ability of body-mounted radiators to reject heat during all phases of the mission. To evaluate the capability of the radiators an analysis was performed to characterize the environment in four locations: low earth orbit, a location 0.5 A.U. from the sun, a location 1.5 A.U. from the sun, and Mars orbit. The resulting sink temperatures are listed below for the four cases:

Table 4. Sink Temperatures for Key Mission Locations

Low Earth orbit (220 nm)	201.6 ^oK (-96.7 ^oF)
0.5 A.U. Heliocentric orbit	222.1 ^oK (-59.9 ^oF)
Mars orbit (220 nm)	163.2 ^oK (-165.8 ^oF)
1.5 A.U. Heliocentric orbit	129.0 ^oK (-227.5 ^oF)

These temperatures indicate that the module will see heat leak in all locations. Radiator size was determined for the warmest case (0.5 A.U. orbit). The results indicate a required area of 78 m^2. This represents 51% of the available area of the cylindrical portion of the shell.

Operations at 1-g would increase coolant pumping losses by ~10% over microgravity conditions, equating to ~100 W of additional pumping power.

Environmental Control and Life Support System

The Air Management Subsystem is characterized by a 4-Bed Molecular Sieve (217.7 kg, 0.6 m^3, 733.9 W), a Sabatier CO_2 Reduction Unit (26 kg, 0.01 m^3, 227.4 W), an Oxygen Generation Subsystem (501 kg, 2.36 m^3, 4,003 W), and high-pressure storage tanks for O_2 (20.4 kg, 0.78 m^3, 6 W) and N_2 (94.4 kg, 3.6 m^3, 6 W). The Water Management Subsystem uses a Vapor Phase Catalytic Ammonia Removal system (1,119 kg, 5.5 m^3, 6,090.7 W) and potable water storage tanks (145.9 kg, 0.54 m^3, 5 W). The Waste Management Subsystem uses a Warm Air Dryer (527.2 kg, 11.2 m^3, 2,043.7 W).

Due to the impact of a 1-g environment on fluid pumping systems, consideration will be given to the placement of the ECLSS pumps such that pumping up and/or down will be gravity-assisted.

Components flown at 1-g could be certified in a ground testbed. Alternatively, construction of an appropriate integrated testbed could be performed on the Earth, thereby alleviating the need to fly the equipment for certification purposes for nominal use conditions. However, systems that are needed and couldn't be shutdown during despun operations would still need to be certified for microgravity operations.

After CO_2 reduction is accomplished in the Sabatier, the stream is passed to a phase separator to separate it into a gaseous stream, which is vented overboard, and a liquid stream, which is sent to the OGS. In the presence of Earth-normal gravity, phase separation could theoretically be accomplished with a settling tank. Knowing that the potential exists for limited exposure to a microgravity environment, this is not a likely design specification; rather, the phase separator would be designed with a centrifugal extraction drum inside of it that tilts along the gravity vector in accordance with the gravitational environment.

In general, a fluid system operating in microgravity will also operate under 1-g with no design changes, especially if this potentiality was noted at the time of vehicle design. There are exceptions to this statement; however, if initial consideration is given to how the gravity vector acts on the system, most aspects of the system can be designed to work in Earth-normal gravity. For example, fans or blowers in a 4BMS drive the stream flow and pumps in a fluid system drive the fluid flow, regardless of the gravitational condition. When sizing the fan or pump, the worst-case scenario will be used.

Two technologies were evaluated for water recovery, the Biological Water Recovery System (BWRS) and the Vapor Phase Catalytic Ammonia Removal (VPCAR) System. A trade study performed determined that the system using the VPCAR, although more power intensive than the BWRS (6,090 W vs. 2,649 W total system power), was preferable due to its lower mass and volume requirements (1,119 kg vs. 1,596 kg mass and 5.5 m^3 vs. 8.0 m^3 volume). As previously mentioned, the longer turnaround time of the BWRS as well as the large BWRS expendable mass (2,703 kg vs. 243 kg) are other disadvantages. Therefore, the VPCAR system was recommended for this vehicle.

In analyzing the CO_2 removal system, the technologies of 4-Bed Molecular Sieve (4BMS) and Solid Amine Vapor Desorption (SAVD) were evaluated. Although the 4BMS was slightly more mass intensive than the SAVD (218 kg vs. 111 kg) and larger in volume (0.6 m^3 vs. 0.2 m^3), its ability to recover H_2O was of value in light of the mission duration. As power is not an issue in the vehicle design, ECLSS elects to use the 4BMS as the CO_2 removal technology.

Consideration was given to warm air drying (WAD) and lyophilization (freeze-drying) solid waste disposal options. While the technologies are similar in mass (527 kg vs. 499 kg) and volume (11.2 m^3 vs. 11.8 m^3), a power comparison demonstrates the more power-intensive nature of the WAD (2,044 W vs. 246 W). Because the technology readiness level (TRL) of the WAD (TRL=8) is expected to remain higher than lyophilization (TRL=5) for the foreseeable future, and based on the longer cycle time of the lyophilization unit, the ECLSS design specifies the WAD technology to process solid waste.

Human Factors and Habitability

The Human Factors and Habitability (HF&H) system includes the galley, wardroom, Waste Collection System (WCS), personal hygiene, clothing, recreational equipment, personal stowage, housekeeping, operational supplies, maintenance, and sleep accommodations.

For the most part, there will be a reduction of complexity in 1-g habitability systems compared with past microgravity spacecraft systems. For example, WCS, personal hygiene systems, and sinks will not need vacuums to control free-floating debris as in microgravity. Also, the galley can be modeled more closely to an Earth-based kitchen with similar types of appliances and food preparation techniques. In order to minimize the amount of consumables required, a dishwasher, clothes washer, and clothes dryer can be incorporated into the design. An additional feature of this habitat as opposed to traditional spacecraft due to the 1-g environment will be the inclusion of beds, chairs, and other Earth-based comfort items. An example floor plan is given in Figure 17. General designs of the systems will typically be simplified by the similarity of requirements to their counterparts used on Earth. This will help expedite the flight certification process.

The power-rich environment will also permit the consideration of items that are power intensive, yet will help to improve the standard of living onboard the spacecraft. For example, appliances such as incinerators, large freezers, microwave ovens, and convection ovens can now be considered.

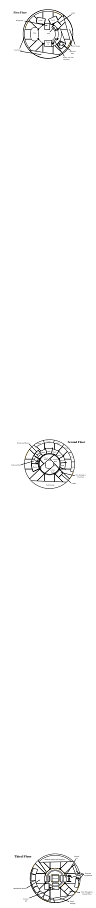


Figure 17. Example Habitat Layout

<u>Systems Issues</u>

Several systems may have significant impacts on the AG vehicle design and operational characteristics, but were not assessed during this study phase. It is expected that subsequent trade studies and integrated design session can aid in understanding their significance.

Many options were discussed regarding techniques for crew ingress and egress from the AG vehicle. It was assumed that during assembly, refit, and resupply operations, the vehicle would be despun and access to the habitat module could be made via a zero-g docking port. However, during mission operations, many options are possible. The simplest would be to despin the vehicle every time the crew must egress or ingress, but this may be quite involved, as the entire vehicle must be "safed" for micro-g operations, and propellant will be expended for each cycle. An alternative would be to provide crew access to the vehicle hub, allowing egress and ingress while under spin.

The transfer of crewmembers to, for example, a Mars lander, provides another set of options. Again, the AG vehicle could despin and the lander docked to the habitat module. Transfer of the crew to the AG vehicle hub and subsequent transfer to the lander by EVA is another possibility. Docking a lander to the AG vehicle under rotation is probably not a good option, as even a docking at the hub would destabilize the rotational motion.

No rigorous assessment was performed of several other systems that may experience added complexity due to the rotating environment. While photovoltaic arrays will not be required on a nuclear powered vehicle, there are several other components that are typically despun or actively pointed. This study assumed that since modern startrackers are essentially electronic cameras with pattern-recognition software, some sort of image compensation algorithms will computationally “despin” the images. It is hoped that some form of phased-array antenna, or switchable fixed beam antennae combined with the high power levels available will enable high-bandwidth communications downlink without a steerable dish. Another method may be required for uplink, however.

Architectural Issues

Several architectural parameters will need to be addressed prior to more detailed assessment, particularly regarding the archetype mission.

In order to ensure the AG vehicle design is feasible from a launch and assembly standpoint, the matters of assembly location (LEO, L_1, etc.) need to be thought out. Also, the transport mechanism of the vehicle or vehicle components to higher Earth orbit should be evaluated, along with the resupply and refurbishment strategy. The infrastructure required to sustain such a reusable vehicle should be also be considered.

It is critically important to understand the destination planetary orbit. While the “minor axis rotation” technique devised for 180° thrust vector shifts can accommodate a certain degree of planetary spiraling, it is not as efficient or fast as conventional tangential thrusting. If routine travel to low planetary orbits is desired, a different vehicle configuration, similar to Figure 1 (Ref. 4) may be a better choice.

Conclusions

The archetype mission requirements were met with a vehicle concept that incorporated acceptable artificial gravity parameters. Additional improvements in transit time and increases in perihelion distance may be possible with more sophisticated trajectory optimization. The vehicle mass associated with the mission is consistent with previous NEP solutions (Ref. 6, 7, 8).

The major challenge unique to the vehicle configuration chosen for this study was met. Steering strategies were identified consistent with the archetype mission requirements without excessive propellant expenditure.

The vehicle mass penalties associated with artificial gravity incorporation appear minimal (a few per cent). The separation distances associated with space nuclear systems were used advantageously to provide the required rotation radius, and the designs for these structures appear to be very lightweight and efficient. No massive despun joints, interface, etc. were required. There was good convergence between the power system mass as the habitat counterweight and propulsive performance utilizing reasonable specific power and thruster performance. Multiple spinup/spindown sequences appear unnecessary, again reducing propellant requirements (although as discussed above, crew egress/ingress techniques are TBD).

Future Work

The system and architecture issues identified above must be addressed. In addition a few targeted studies similar to those presented in the attachments may be desirable. A more detailed power system and main radiator design would be of interest, along with radiator construction or deployment strategies. Reactor radiation scattering and shielding assessments would be needed to validate the overall vehicle configuration.

Finally, a more thorough understanding of the forces and moments the AG vehicle will experience while in a despun mode is required. Docking loads, plume impingement forces, maximum maneuvering requirements, etc. may be more significant structural design drivers than the loads identified during AG operations.

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Attachments

Attachment 1

Reference Structural Configuration Development

Attachment 2

[Artificial Gravity Habitat Phase I - Final Report]

Attachment 3

Artificial Gravity Dynamics Assessment