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Fusion Powered Human Transport to Mars (UWFR94)

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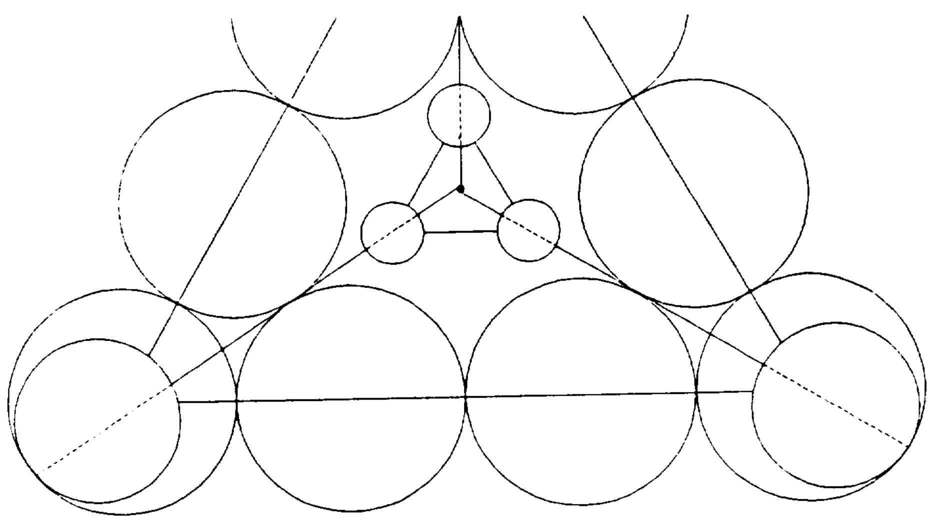

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Abstract

In the future, two important technological dreams will have become reality: fusion will be a viable power source, and human settlements on Mars will be feasible, desirable, and even necessary. Merging these two concepts is especially attractive for the aerospace engineer because of the high specific power that will be possible with fusion (on the order 10 kW/kg).

The UWFR94, a large, fusion-powered, human-transport ship, is designed to transport 100 passengers between Earth and Mars in approximately thirty days. This relatively short transit time, which mitigates the need for artificial gravity, is made possible by a Polywell™ inertial-electrostatic fusion reactor capable of 20 kW/kg. The mass of each reactor is 37 metric tons and the fuel used is ^3He-^3He. The electricity generated drives the propulsion system, composed of nine ion thrusters and 780 tons of xenon propellant. The payload consists of three independent, identical cylinders housing the crew, and has a mass of approximately 400 tons. The aluminum cylinders' radius and length are 3 and 12 meters, respectively, with a thickness of 6 cm (15 cm in the solar flare safe rooms). Atmospheric re-entry is avoided by constructing and repairing the UWFR94 in space, and by transferring crew and cargo to shuttle-like vehicles for transportation to the planet upon arrival.

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Introduction

In the future, two important technological dreams will have become reality. First, human settlements on Mars will be feasible, desirable, and even necessary, as research outposts become mining colonies, and mining colonies develop into thriving communities. Second, fusion will become a viable power source. Merging these two concepts is especially attractive for the aerospace engineer because of the high specific power (on the order 10 kW/kg) that will be possible with fusion.

The UWFR94 (see Figure 1 at the end of the Introduction) is a large, fusion-powered, human-transport ship, designed to transport one hundred passengers between Earth and Mars in approximately 30 days. It will travel 9.74×10^7 km between a maintenance station at Lagrange Point 2 (off the far side of the moon) and a parking orbit about Phobos, the inner moon of Mars. The UWFR94 will achieve a maximum velocity of 90 km/s ten days into the mission, after which it will "glide" for ten days as it rotates 180 degrees to begin slowing to its destination. Atmospheric re-entry is avoided by constructing, repairing and refueling the UWFR94 in space, and by transferring crew and cargo to shuttle-like vehicles for transportation to the planet upon arrival.

The relatively short transit time, which attenuates the need for artificial gravity, is made possible by three Polywell™ inertial-electrostatic fusion reactors capable of 20 kW/kg. Each 37-ton reactor is 8 meters in radius and uses ^3He-^3He as fuel, which produces only a minor amount of bremsstrahlung radiation. The electricity generated drives the propulsion system, composed of nine 11-meter-radius ion thrusters and 780 tons of xenon propellant.

The 400-ton payload mass consists of three independent cylindrical modules, housing the crew in a volume of more than 1300 m^3. The identical, spherically-capped cylinders, made of Aluminum-2024, have a radius of 3 meters, a length of 12 meters, and a thickness of 6 cm (15 cm in the solar flare safe rooms) to ensure that the total radiation dose received over the course of the mission is not more than 5 Rem. The interior of the modules is divided into 6 floors of living space, and has an environment of 70% nitrogen - 30% oxygen pressurized to one atmosphere.

A complete list of design specifications may be found in Appendix A.1.

UWFR94 Structural Design

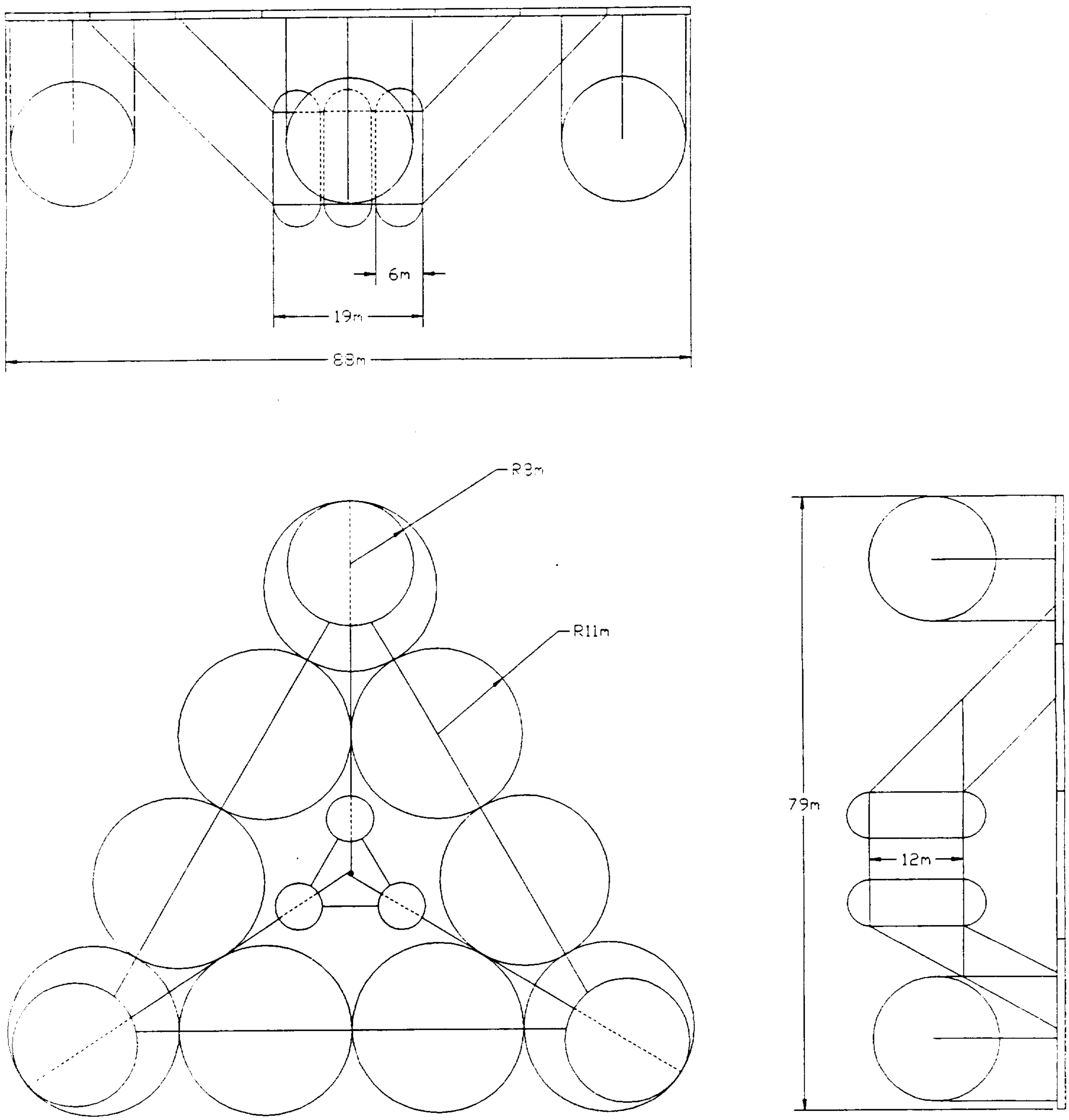

FIGURE 1

Design Process

The 1994 Fusion Rocket Team was unique among the Engineering Mechanics senior design groups. Our team chose the mission for which the UWFR94 was designed. In this chapter the design process is briefly described.

Mission Definition

The team was introduced to the inertial-electrostatic fusion reactor and its performance characteristics. From there, the mission – destination, payload, transit time, and so on – was defined on the basis of the research and analysis interests of the team members. After discussing options with Mr. Thomson and Dr. Santarius and reviewing the work of previous fusion rocket groups, the team made the decision to design for a short duration mission to Mars with a crew of approximately 100. This was based on the following observations and opinions.

- There was a great deal of interest in diverging from the course the previous fusion teams had taken. For example, designing for artificial gravity would be avoided if possible. Similarly, if ^3He-^3He could be used as the reactor fuel, shielding to protect the spacecraft and its inhabitants from neutron radiation from the reactor would not be required.

- It was felt that certain fundamental areas had been neglected in previous reports. For instance, the exterior material for the habitation modules had never been thoroughly discussed.

- Mars is a suitable destination due to its proximity to Earth, which, combined with the power characteristics of the reactor, made a short duration mission feasible.

- Without the mass penalties of an artificial gravity system and an enormous reactor shield, a much larger crew is possible.

- A manned mission to Mars has been the subject of intense research recently, so the relevant literature is abundant.

Preliminary Work

Next the team constructed a needs analysis for the mission; it may be found in Appendix A.2. This helped to delineate preliminary research areas and suggest potential ship configurations. Hence, research was begun on the following topics.

- trajectory optimization
- reactor fuel selection- ^{3}He-^{3}He feasibility
- thruster and propellant selection
- exterior surface material selection
- space environment considerations
- dimension optimization
- life support

The process of selecting the UWFR94's configuration proceeded with a number of creativity and critical evaluation meetings. A prefatory PERT diagram was also created, outlining the team's plan for the semester. This outline was updated frequently in the early stages as the scope of the project came into focus. The team's final PERT is found in App. A.3.

Subsequent Work

When the team completed the preliminary research and selected a configuration, a second phase of research and analysis began, continuing with earlier work and undertaking new tasks in the following areas.

- FAST Diagram (App. A.4)
  The FAST Diagram helped in understanding the primary requirements of the design.
- interconnecting supports
- assembly considerations
- heat transfer analysis
- interior design
- waste management

The configuration changed slightly upon completion of the trajectory analysis with the addition of two more reactors for a total of three. Likewise, the ion thrusters had to be modified to reduce size (while maintaining feasibility). In general this stage of the design was marked by an increased interdependence among the team's research groups.

Finally, the team compiled the work and wrote the final report.

Chapter 1
Getting There

§ 1.1 THE POLYWELL–AN INERTIAL-ELECTROSTATIC REACTOR

An Overview

The subject of fusion may be foreign to many people. This is an introduction of how fusion the fusion process occurs, and how fusion power is converted to electricity using the Polywell device. Dimensions, fuel choice and materials will be discussed in the next section.

Below is a two-dimensional representation of the Polywell reactor.

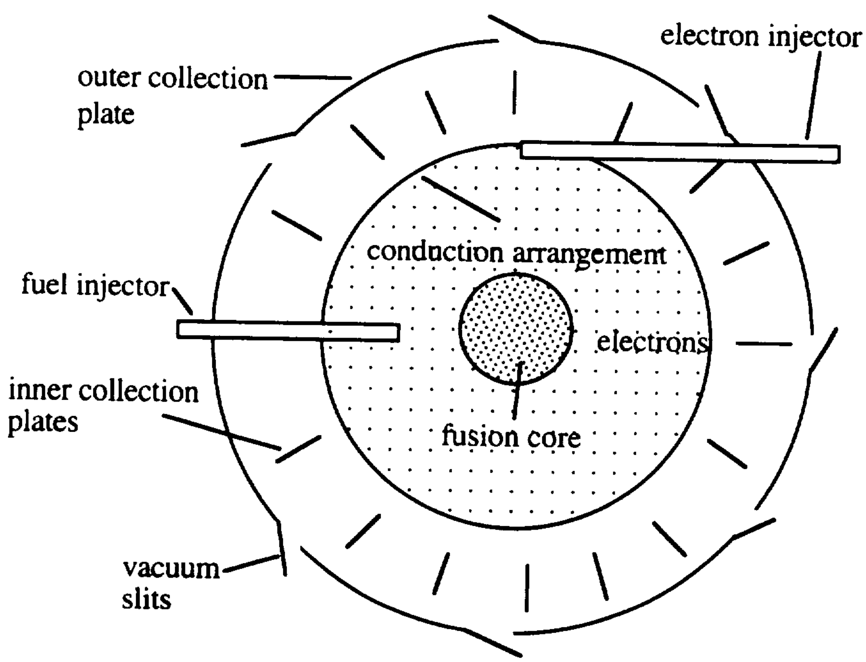

FIGURE 1. The Polywell.

The negative potential well

The Polywell uses the negative potential well to confine the fusion fuels in a dense plasma core in the center of the sphere, where the fusion reactions take place.

To create a negative potential well, a truncated cube conductor arrangement can be used as shown below.

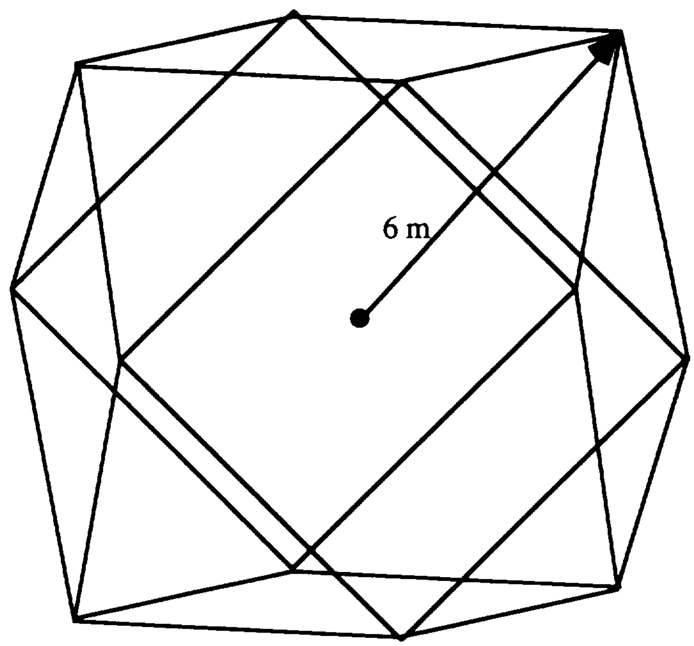

FIGURE 2. The Conduction Arrangement [1].

A current runs around each triangular segment creating a magnetic field that almost completely surrounds the inside of the conductor arrangement. The magnetic field is strongest at the conductor arrangement, and decreases radially until it becomes zero at the center of the reactor [2].

Electrons and the fuel are then injected into the Polywell at the location where the magnetic field is the strongest [1]. The fuel is immediately ionized and its electrons join the ones that are injected [3]. The electrons are reflected outward from the weaker magnetic field, which is at a lesser radius, to the stronger magnetic field, closer to the conductor arrangement, if the electrons are traveling at a velocity perpendicular to the magnetic field. Hence, the electrons surround the inside of the conductor arrangement in spherical fashion. These electrons in this pattern form an electric field that, like the magnetic field, is strongest at the conductor arrangement, and nonexistent at the center of the reactor [2].

The electric field causes the newly formed fuel ions to fall to the center of the reactor [3]. The electron injector insures that there will always be an excess of negative charge (electrons) as compared to positive charge (fuel ions). There would be no well if there was no charge disparity, because the ions would neutralize the electric field.

The spherical electric field can be made analogous to a spherical gravitational field, such as the Earth. Just like this electric field, the Earth's gravitational field is the strongest on the outside, and zero in the center. To continue the analogy, we shall say that the Earth has a hole going all the way through it, as shown below.

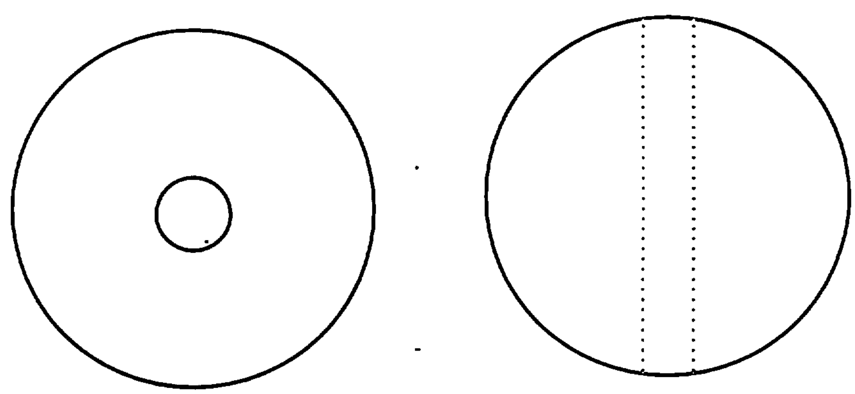

FIGURE 3. Earth With a Hole.

(Keep in mind that the gravitational pull of the sphere at a particular radius down the hole depends only upon the mass from the center of the sphere to the radius at which the object is located, and does not depend on any of the mass of the sphere above that radius.)

An object is dropped into the hole at the surface, and it accelerates quickly at first (9.81 m/s^2) but the acceleration decreases as it continues down to the center. At or near the center, the object is traveling at almost a constant speed, and its momentum carries it through the center at its top speed. It then goes up the hole on the other side of the center, and decelerates until it reaches zero velocity at the surface. The object will oscillate through the center in this manner (law of conservation of energy).

This is very much what happens with the ion and the electric field. The difference between a spherical gravitational field and a spherical electric field is that the force subjected on the ion by the electric field at a given radius is dependent upon the field *above* the radius at which the ion is, and not at all dependent upon the field between this radius and the center. Both fields, however, accelerate their particles to the center–with decreasing acceleration– and cause the particle to oscillate. This oscillation is known as the orbit of the fuel ion [3].

All ions will have similar radial orbits that travel through the center of the sphere (the fusion core), but the orbits will originate from different points along the conduction arrangement. Electrons must be continuously injected because some of them escape through the magnetic field [2].

The fusion reaction

As stated previously, the ions oscillate though the center of the sphere; this is the only point where their orbits intersect. When two or more ions collide, they may simply bounce off each other, but sometimes a fusion reaction takes place. The fuel breaks apart at the subatomic level. The bonds that hold the protons and neutrons in the nucleus of the atom are broken and this bond energy is converted to kinetic energy of the fusion products [3]. Further explanation of the fusion reaction of the chosen fuel will be given later.

Conversion to electricity

Since there is a net kinetic energy in the fusion reaction, the products will leave the center of the sphere with a greater velocity than the reactants had just before the collision. This means that the fusion products will have enough velocity to move beyond the electric field, the sea of electrons, the magnetic field and conduction arrangement [2]. Ion collection plates are set up beyond the conduction arrangement [5], as is shown in Figure 1.

After the fusion reaction occurs, the speeding particles head towards the collection plates. The outer sphere also acts as a collection plate. The collection plates are placed at radii where the particles are known to have zero kinetic energy after they decelerate. The further away from the center the ion is collected, the more electricity is created. Below is a diagram of the circuit representation of the conversion to electricity.

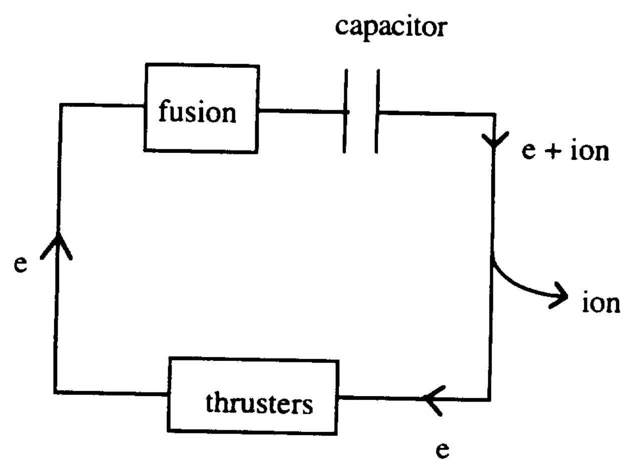

FIGURE 4. Circuit Representation.

After the fusion reaction, a newly formed ion moves radially outward. The ion is trailed by electrons due to coulombic forces; these electrons "want" to neutralize the ion [3]. As the ion moves outward, its velocity is decreasing--this energy is being converted to electricity and stored in a "central capacitor". The circuit, though, can only be completed if the ion enters the collection plate, followed by the trail of electrons. When this is done, the capacitor can release the current which would then go to the thrusters or any place needed. Once the circuit is complete, the ion is immediately neutralized. The neutralized ion travels with the electrons in the current until it is discharged [3].

Due to the abundance of electrons surrounding the fusion core, no wire is needed to carry a current between the core and the collection plate. If an ion does not enter a plate, it will accelerate back down to the fusion core, draining the capacitor of all the energy with which it had initially charged it up.

One inefficiency occurs if an ion enters the collection plate with some kinetic energy left. This kinetic energy is transformed into thermal energy, and is radiated away [3]. Sometimes an ion's energy will carry it a little distance past a collection plate. The ion will move around transversely until it dissipates enough energy (draining the central capacitor) to enter a collection plate below it. The following is an illustration of this.

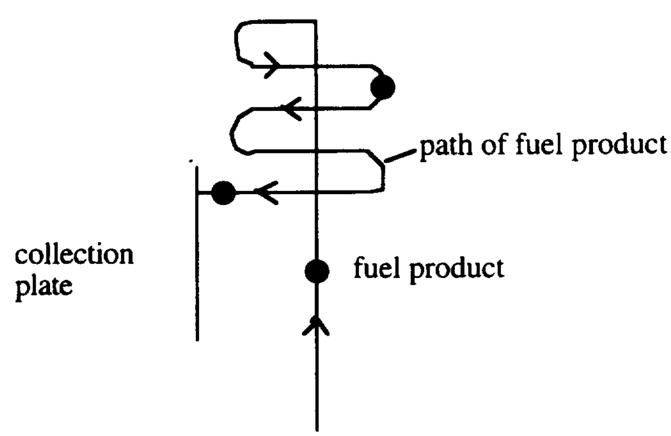

FIGURE 5. The Ion Dissipating.

Another inefficiency is the electric field slowing the particles down as they radiate outward past the conduction arrangement [2]. Taking these two into account, the efficiency of the system is still very good.

Radiation

In fusion there are two kinds of radiation: synchrotron and bremsstrahlung. Synchrotron radiation is the emittance of infrared radiation [3] due to particles moving perpendicular to a magnetic field [4]. In the Polywell, the fusion reaction occurs in or near the center of the sphere, where the magnetic field is nearly nonexistent, therefore there is no synchrotronic radiation from the fusion [2]. Some synchrotron radiation is cause by the electrons moving through the magnetic field, but it is negligible.

Bremsstrahlung radiation–accelerated particles releasing x-rays–is more severe than synchrotronic, but is not powerful enough to reach the passengers outside the outer collection shell.

Vacuum system

A low pressure of about 10^{-3} torr is needed in the volume between the conducting arrangement and the outer collection shell [3]. This pressure can be maintained by utilizing the vacuum of space. Slits must be made in the outer shell, and the required amount of neutral particles are blown out. These slits are shown in Figure 1. If there are too many neutral particles floating around in the reactor, they may collide with high energy ions to produce fast neutral particles, which cannot be contained and do not produce any electricity, and random directional ions, making it difficult to harness their energy.

Cooling

The only item that may need to be cooled is the conduction arrangement that produces the magnetic field. All other pieces can radiate the thermal energy away [5]. A probable coolant is ^4He [3], or water [3].

The Polywell is a unique way to accomplish fusion. The low radiation levels and its ability to directly convert fusion energy to electricity make it very appealing for space travel.

Specifications

A Polywell fusion device is very complicated, so only four facets of it were designed or chosen: the fuel, the magnet system, the outer collection shell, and attaching the engine to the thrusters.

Choosing a fuel

The fusion process revolves around many important variables, but perhaps the most important is the type of fuel.

Desired Properties of Fuel- Reaction must have a high energy release

Undesired Properties- Reaction must not contain n, T, or D

n (neutron) - It has no charge, therefore it cannot be contained by the magnetic field that surrounds the reaction. This contributes to the breakdown of the structure.

T (tritium (Hydrogen with two neutrons)) - Very radioactive, therefore more shielding is needed; dangerous to passengers.

D (deuterium (Hydrogen with one neutron)) - Very reactive--the product of D + D yields tritium in some cases. SOME AMOUNT OF D IS ALLOWABLE.

1) All reactions (fuels) with energy release above 8 MeV were chosen from a list in a report by McNally [4].

a. D + T --> n + ^{4}He (17.586 MeV)
b. D + ^{3}He --> p + ^{4}He (18.341)
c. D + ^{6}Li --> 2^{4}He (22.374)
d. T + T --> 2n + ^{4}He (11.327)
e. T + ^{3}He --> D + ^{4}He (14.319)
f. ^{3}He + ^{3}He --> 2p + ^{4}He (12.861)
g. p + ^{11}B --> 3^{4}He (8.664)
h. ^{3}He + ^{6}Li -->p + 2^{4}He (16.880)

2) Reactions with n and T eliminated outright.

a, d and e eliminated.

3) The remaining five are analyzed separately

b- ^{3}He hard to find and/or develop
Some D with react with itself to form T
Not very much radiation, but some

c- Large energy release
very complicated reaction
Lithium cools reaction, increasing plasma density
Severe solid ash problems in high vacuum systems (unburned fuel)

f- ^{3}He not available
low reactivity
^{3}He has nuclear elastic collisions-- improve reaction
no products of D, n or T
fuel is non condensable, so no ash problem

g- small energy release
low amount of n produced

secondary reaction

h- reaction not likely to occur, need P + ^{6}Li --> ^{3}He + ..., then ^{3}He must react secondarily with ^{6}Li. This only happens 10-20% of the time.

4) Weed out new undesirables

c- ash problems, contains some deuterium

h- would need too much ^{6}Li fuel because of the 10-20% chance of reaction

b- some deuterium plus the rarity of ^{3}He makes bad choice

5) Must decide between g and f

- g has definite energy from products, unlike f
  f has low reactivity, and fuel availability problem
  f has a higher net energy

Conclusion : Both ^{3}He + ^{3}He and p + ^{11}B would be ideal for our engine. We chose ^{3}He + ^{3}He because we are transporting 100 people and the higher energy fuel would be more efficient.

The Fusion Reaction for ^{3}He Fuel

The following is the fusion reaction of two ^{3}He particles shown is equation form and in pictorial form.

$$^3\text{He} \quad + \quad ^3\text{He} \quad \text{------>} \quad 2p \quad + \quad ^4\text{He} \quad + 12.861 \text{ MeV [4]}$$

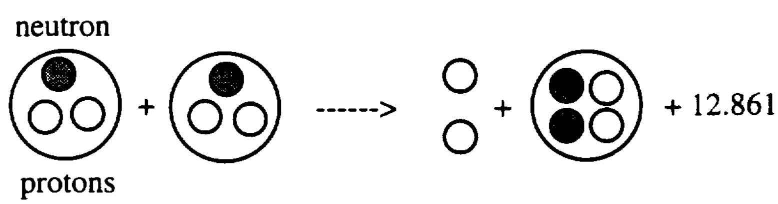

FIGURE 6. Fusion reaction for ^{3}He+^{3}He.

As can be seen from the pictorial form, there are an equal number of particles on each side of the equation. But if the left side and the right side were both weighed, the mass of the two ^{3}He particles would be greater than the mass of 2p and ^{4}He. The difference of this mass corresponds to an energy of 12.861 MeV utilizing the equation $E=mc^2$, where m is the difference of mass. This equation implies that the strong nuclear forces that hold the protons and neutrons inside the nucleus of the ^{3}He ions actually have some mass. The energy from the strong nuclear forces is converted to kinetic energy of the reaction products during the fusion reaction. This energy must then be harnessed and converted into electricity.

After the fuel and trajectory (later chapter) were given to our advisor, the following numbers were provided [8].

1.	Number of reactors needed	3
2.	Electric power to the thrusters	2910 MWe
3.	Gross electric power	3170 MWe
4.	Gross fusion power	3960 MW
5.	Electric power to injectors	260 MWe
6.	Input power to plasma	230 MW
7.	Heat that must be radiated	790 MW
8.	Direct converter efficiency	.8
9.	Power injector efficiency	.9
10.	Plasma radius	5 m
11.	Radius to magnets	6 m
12.	Radius to last direct converter	8 m
13.	Heat flux on magnet inner face	8.8 MW/m^2
14.	Heat flux on last direct converter plate	1.1 MW/m^2
15.	Magnetic field	1 T
16.	Radius of convergence	.005 m
17.	Area fraction intercepted by magnets	.03

Parameters (1-17) are per reactor.

Items 2 - 9 are illustrated by the following flow chart.

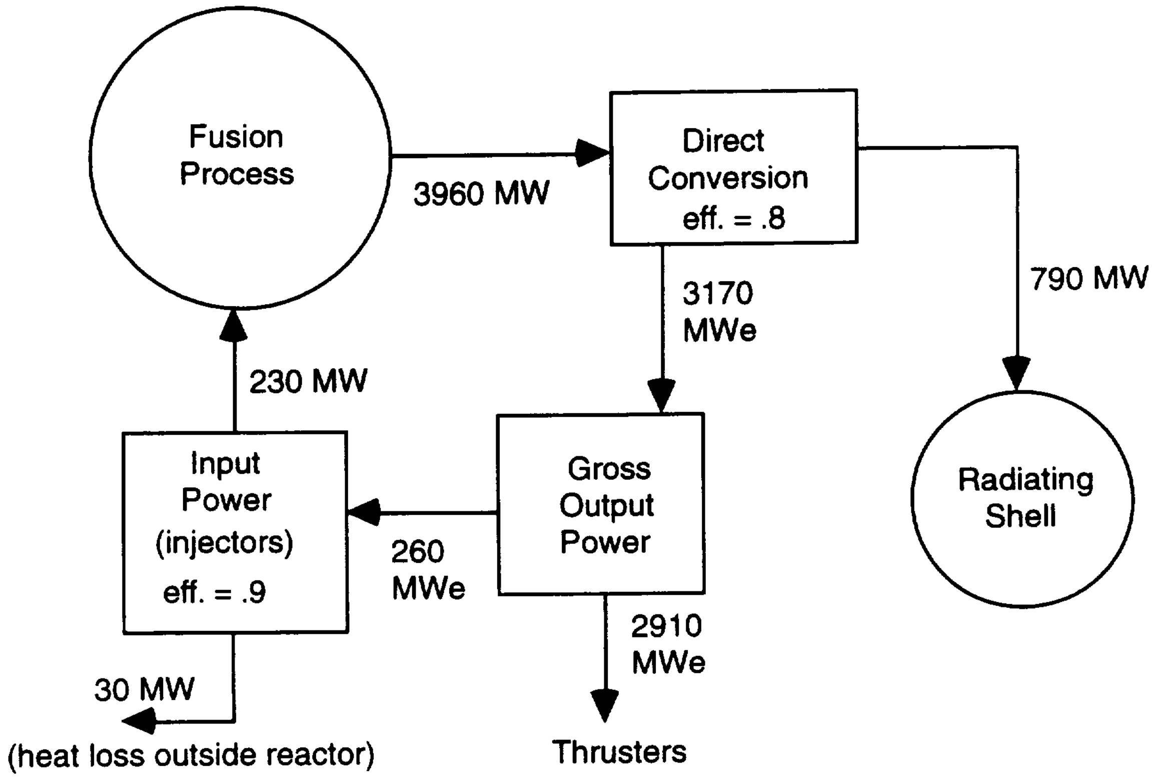

FIGURE 7 . Flow chart of fusion power (per reactor).

230 MW of power is injected inside the conduction arrangement (magnet) if the form of speeding electrons. See App. B.1.1 for the energy equation. These speeding electrons develop the negative potential well, and the fusion process follows. 3960 MW are created by this fusion reaction. 80% of this energy is converted to electricity, while 20% is lost to heat energy and must be radiated away from the reactor. Of the 3170 MWe of electricity, 2910 MWe go to the thrusters, and 260 MWe go to the electron injectors, which have a 90% efficiency.

The place to which all these numbers were rounded is greater than the total operating power of all the other ships systems (everything except the thrusters and the engine), and therefore they were not included in the flow chart.

Items 10-16 are illustrated below.

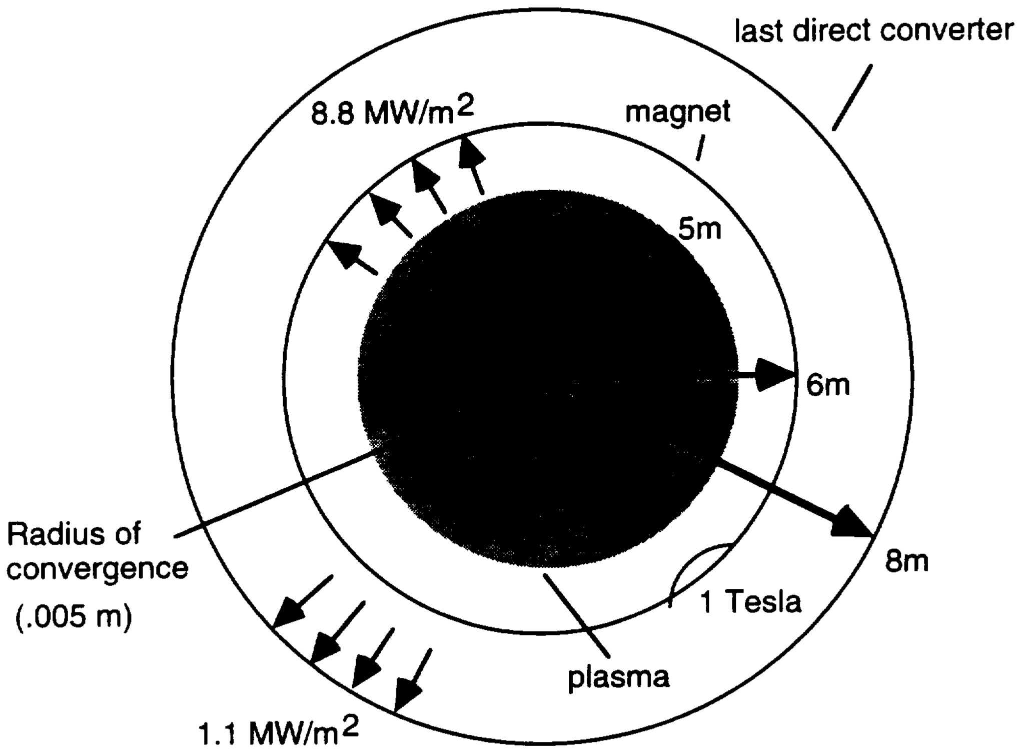

FIGURE 8. The engine and some of its characteristics.

One must remember that this is a two-dimensional representation of a three-dimensional unit. All circles shown are actually spheres. The sphere that the radius of convergence outlines is where all the fusion reactions take place.

The last direct converted, or the outer shell, is used to dissipate heat from the engine to outer space. We chose the shell to be made of Tungsten, with a thin coat of Carbon lining the outside wall. Carbon was used because of its high melting point, and very high emissivity. Tungsten was chosen because of its high melting point (3673 K), and it is generally good electrical conductor. No other materials could be found that had these properties. App.

B.1.2 shows that the melting points and the emissivity are adequate for the design. Next is an illustration.

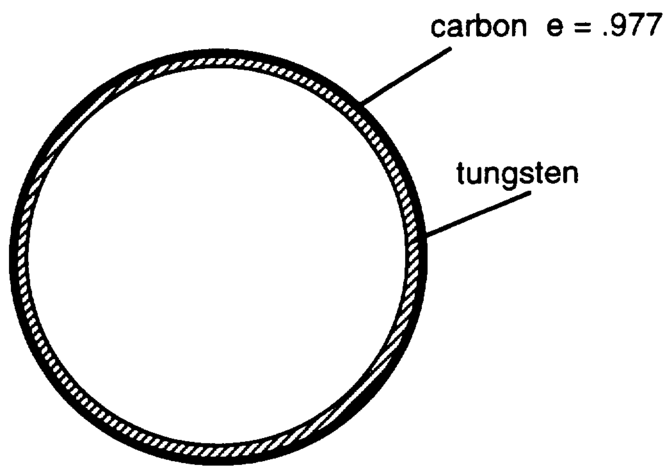

FIGURE 9. The outer converter (shell).

The Conduction Arrangement (Magnet)

The magnet is shown to have 8.8 MW/m^2 of heat incident upon it. This would cause it to have a very large temperature. Therefore, a layer of carbon was place around each wire. The diagram is shown below. The diameter of the wire is 19 cm. The calculations are in App. B.1.3.

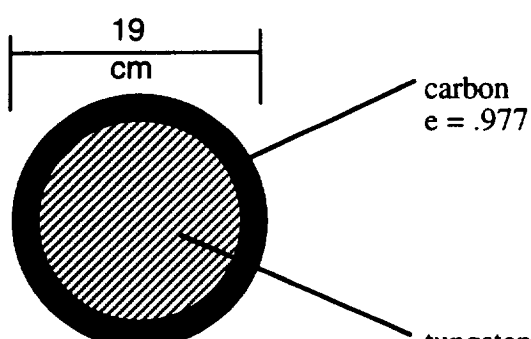

FIGURE 10. Cross section of magnet wire and carbon layer.

The layer of carbon is very thin. The drawing is exaggerated so the components can be seen more clearly. This design will allow the tungsten wire to remain cooler despite the heat flux. The current through the tungsten may have to be increased to make a one Tesla magnetic field, because the carbon layer may block some of the field.

Attaching the Engine to the Thrusters

The only force that will be acting on the engine is the force due to acceleration. This force was calculated to be 373.7 N, is explained in App. B.1.4. The setup for attachment is illustrated below.

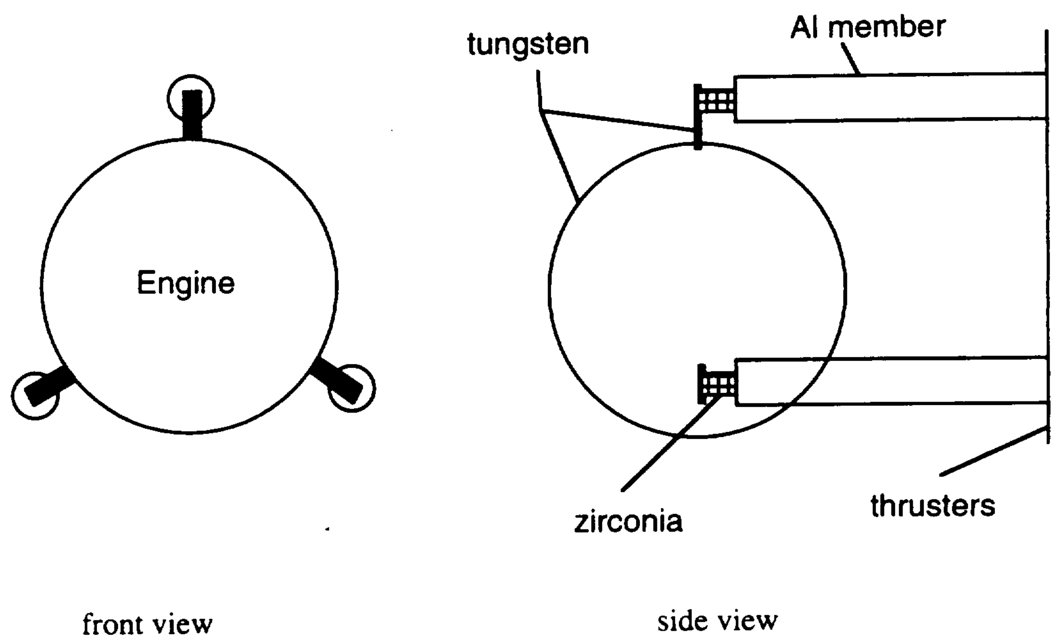

FIGURE 11. Attaching the engine to the thrusters.

The aluminum members, with a melting point of about 1000 K, would melt if they were attached directly to the engine. Therefore, circular zirconia bars are attached at the ends of the aluminum members. The following diagram shows the specific dimensions. The calculations are in the App. B.1.5.

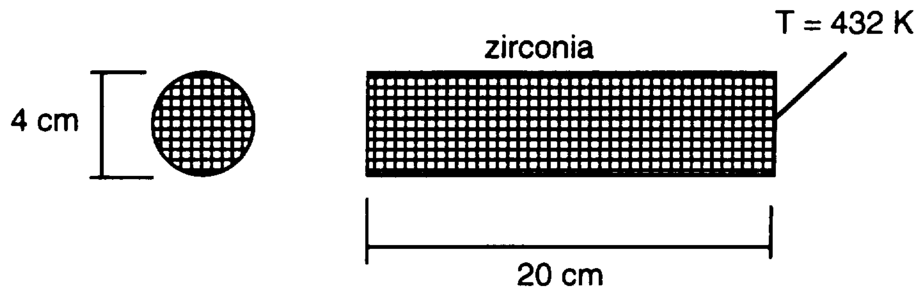

FIGURE 12. Zirconia bar.

At such a high temperature, zirconia only has strength in compression. Therefore, tungsten plates protrude out of the engine, where the zirconia can be attached, as shown in the Figure 11. While the craft accelerates or decelerates, there will always be a compression on the zirconia. App. B.1.6 has the calculation that the zirconia can handle the acceleration forces.

The aluminum bars have to be designed next. They will be hollow aluminum cylinders. The aluminum is used because of its high strength to weight ratio. These cylinders will buckle before they will fail in compression. The thickness was set at .16 cm, and the outer radius came out to be 4.9 cm. This is explained in App. B.1.7.

Materials such as carbon and zirconia may be hard to shape, but because of the high temperatures, they are necessary.

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§ 1.2 ELECTROSTATIC THRUSTERS AND PROPELLANT

In this section, the thruster system will be discussed. We will be using ion thrusters, also known as electrostatic thrusters, to propel the craft to Mars. First, there will be discussion of ion thruster functions, basic styles of ion thrusters, special features that can be added to ion thrusters, propellant selection and propellant storage. Then, the ion thrusters that will be used for UWFR94 will be discussed.

The Basics of the Ion Thruster

The ion thruster can be broken down into three major parts: the ion source, the acceleration and the neutralization. First the ion source will be discussed.

Ion Source

The surface contact source consists of a metal plate and a propellant vapor. The propellant vapor passes through an ionizer [11]. A common combination for this process is cesium and tungsten. Fig. 1 [1] below, shows the process of a ion surface contact source.

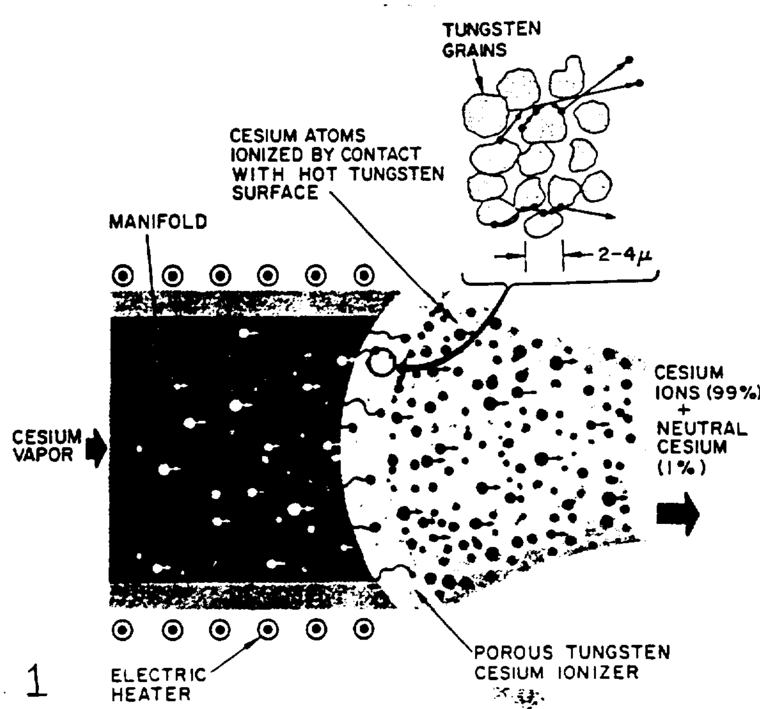


FIG. 1

Electron bombardment is the process where electrons oscillate through a beam of neutral atoms until they lose their energy in ionizing collisions. [11] Fig. 2 [4] below, shows a schematic for an electron bombardment type ion thruster.

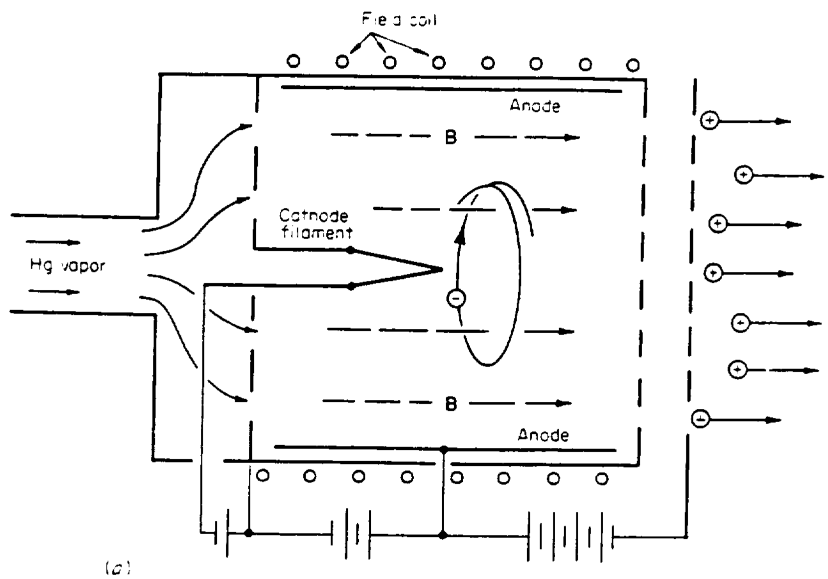


FIGURE 2

Acceleration

To accelerate the ions, the thruster uses a voltage potential to set up a current between two plates. The current created is the flow of the ions. There is a limit on this flow called the space charge limit. This limit follows the Child- Langmuir Law [11]. The space charge depends on the voltage potential, the distance between the plates and the charge to mass ratio of the propellant. The voltage breakdown point (or voltage arcing) puts a limit on the distance between the plates [11].

The space charge law: $i = 4E_0/9\ (2\varepsilon/\mu)^{1/2}\ (U^{3/2} / s^2)$

where: $E_0 = 1/36\pi$ e-19 (amp sec/volt m)

s = distance between plates

ε = charge of particle

μ = mass

U = voltage

The ion beam is focused with electrodes to help prevent sputtering. Sputtering occurs when stray ions bounce around causing damage to the thruster.

Neutralization

Neutralization of the ion beam is extremely important. To neutralize the beam, oppositely charged electrons are added to the ion beam. Adding these particles keeps the ship from building up a potential. If exhaust were not neutralized, ions would turn around in the potential field and impact the space vehicle[11].

There are four main ways to neutralize the beam: thermionic emitter near the beam, thermionic emitter in the beam, electron gun or plasma bridge [9]. Fig. 3 [9] below, shows a schematic of each of these methods respectively. Neutralization, as a rule of thumb, should be done at a distance of 2d from the exit point (where d represents the separation width the acceleration grid) [9].

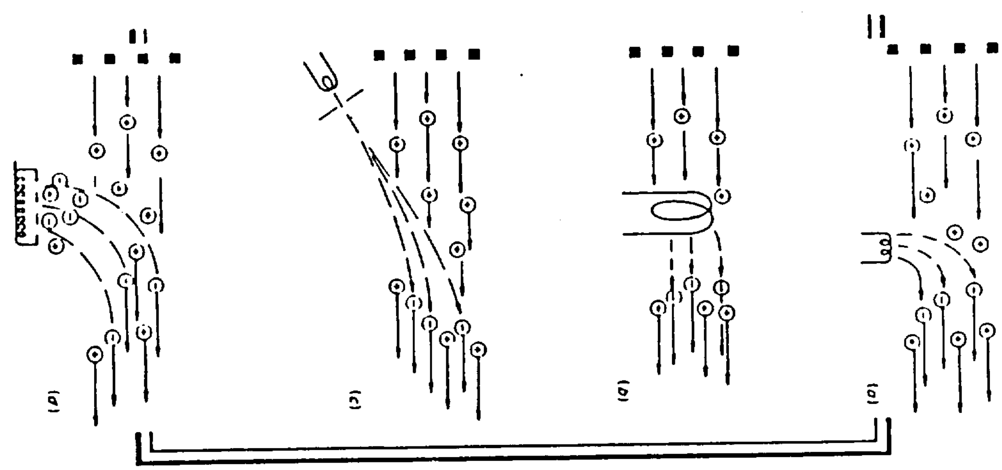


FIGURE 3

Thruster Styles

Basic Design

There are three basic source styles of the ion thruster: the planer, where the ion source is the same size as the final beam; the cylindrically convergent, which transmits a linear beam strip from a larger source; and the spherically convergent, which transmits a circular beam from a larger source. The spherically convergent style can be used for both contact and electrical bombardment ion sources. This configuration makes it possible to design a compact and efficient ion source because it allows for space between the beams. This space makes it possible for the structure to be rigid enough to withstand erosion of the acceleration and deceleration electrodes for a long period of time [1]. Fig.4 [1] below, shows a schematic drawing for each of the previously described styles.

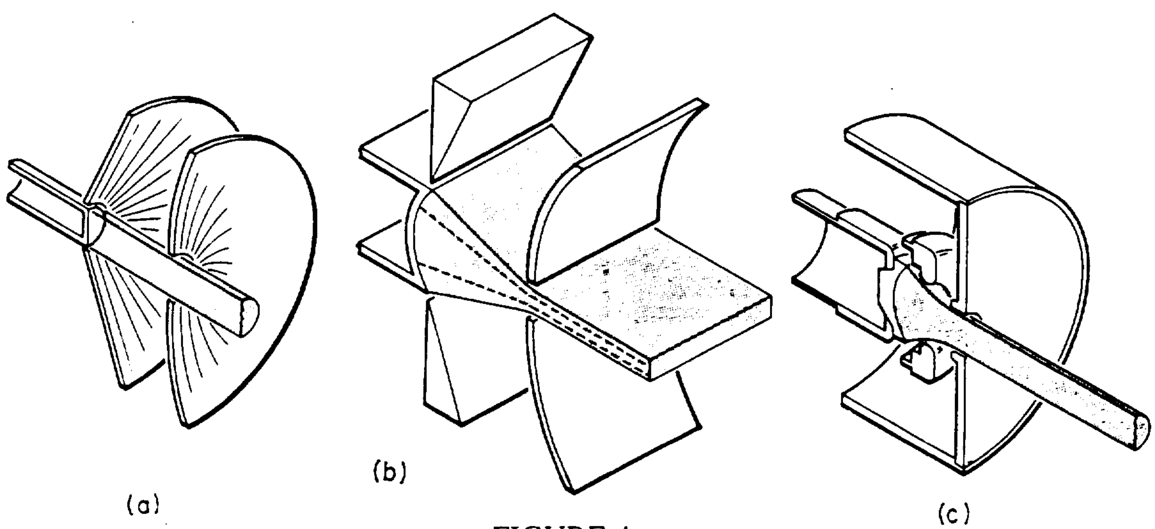


FIGURE 4
(a) planer (b) cylindrically convergent (c) spherically convergent

Acceleration grids

There are two types of acceleration grids: the button configuration and the linear configuration. The button configuration is used in multi-source units and it is commonly used for contact or electron bombardment sources. The linear configuration is usually used for a single source but can be used for multiple sources [1]. Fig.5 and Fig.6 [1] below and on the following page, show the above configurations used separately as well as multiple source accelerators.

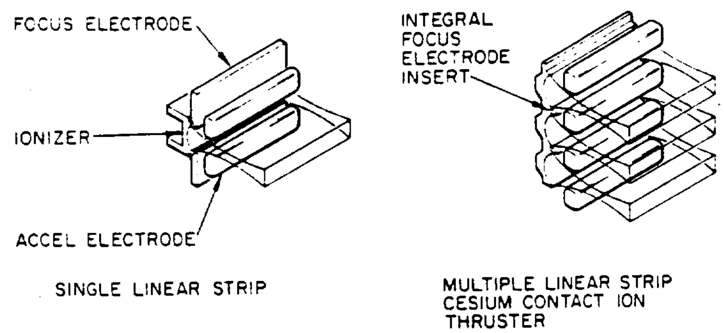


FIGURE 5 Linear Sources

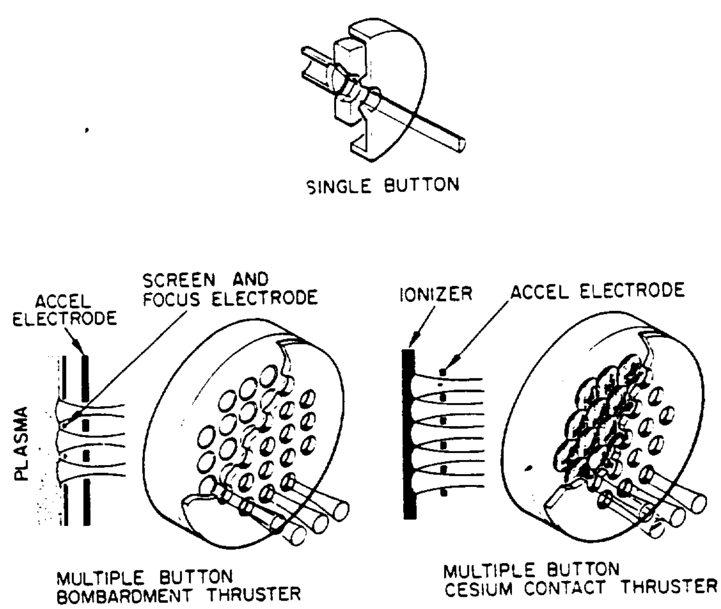


FIGURE 6 Button Sources

Beam spreading

The ion beam, after leaving the acceleration grid, will start to spread. The more the beam spreads the less net thrust produced by the thruster. The angle of deflection for a spreading beam from the horizontal follows [11]:

$$\tan\Theta = 0.48R$$

where: $R=2r/s$

r= radius of thruster

s= length of thruster

Steering using Electric fields

Steering using electric fields is a unique feature for ion thrusters and creates a definite advantage over other types of thrusters. The net charge on the ion beam is positive, therefore a non symmetrical magnetic field on the accelerator would create a net deflection [1]. The angle at which the beam is deflected follows this relation:

$\tan\theta=(\Delta Va/U)(L/t)$.

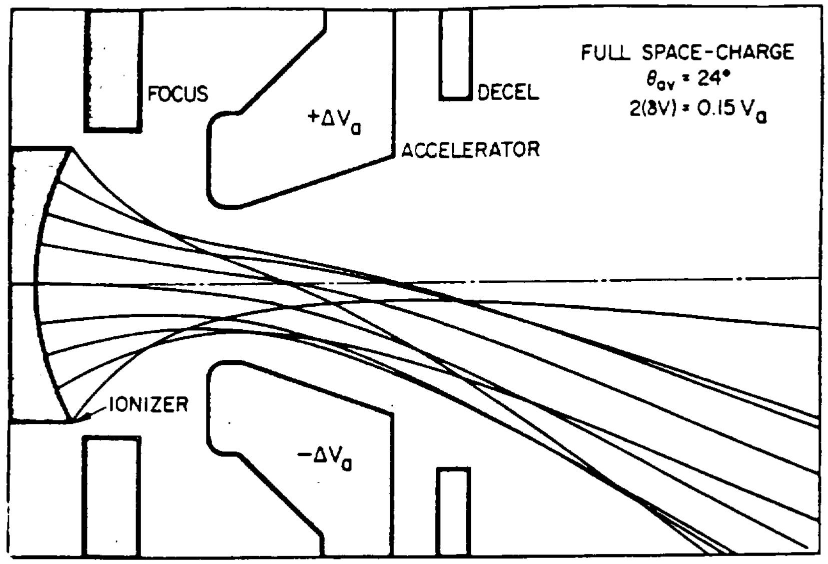
ΔVa = voltage difference applied to cause deflection,

U is the voltage related to the exit velocity,

L is the length that ΔVa is applied over (the deflection electrode)

t refers to the width in which the ion is released

A graph relating the defection to the normalized voltage for cylindrical and linear acceleration grid is in Appendix B.2.1 [1]. Fig. 7 on the following page, [1] demonstrates how deflection is done.



Deflected ion beam trajectories as determined in an automatic trajectory tracer for a linear strip ion thruster including the effect of space charge.

FIGURE 7

Propellant Selection

The main things to keep in mind when picking a propellant are low melting temperatures, low vaporization temperature, ease of ionization (low ionization potential) and corrosion problems with the tank, pipes, valves, heater and gages. The mass to charge ratio of the propellant should also be high [11].

It is to our advantage to use a large particle propellant [11]. Even though this type of propellant weighs more then a small particle propellant, we can design a lighter thruster. The current density determines the area of the thruster. The heavier propellants can have a higher current density, therefore reducing the area needed. Besides the advantage of the current density, large particle propellant operates at a high voltage and a low current whereas a small particle propellant operates at a low voltage and a high current [9]. The lower the current, the smaller your area. Therefore, by using a large particle propellant it is possible to design a lighter thruster.

In determining the total mass, it is the size of the thruster and storage tanks that will change the mass of the system. The mass of the propellant will remain constant. This is because the mass of the propellant is determined by the mass flow rate required for the thruster. This mass flow rate depends only on the total power supplied to the thruster and the exit velocity desired, so the mass of the propellant needed will remain constant no matter what propellant is chosen.

Cesium and mercury meet these requirements and both were used for early experiments. Problems with erosion and health hazards have moved experiments away from these two elements and onto the inert gasses. Of the inert gasses, xenon and krypton have been used the most in modern day experiments. Hydrogen, argon, and neon have also been considered, but little experimental work has been done using these as propellants.

Propellant Storage

There are two main ways in which propellant can be stored. The propellant can be stored in a contraction tank which contracts to create a mass flow of the liquid propellant. The liquid propellant passes through a heater where it becomes a vapor. Another way to store the propellant is to store it in a pressurizes tank that is heated to create vapor. The flow of the propellant vapor would be controlled by valves. To separate the vapor from the liquid in weightlessness the tank needs to rotate. If the ship is accelerating at least 1e10-4g_o then rotation is not necessary for separation.[11]

Thruster Selection

Propellant Choice

The propellant chosen is xenon. The mass estimate for the thrusters was the lowest for this propellant. Krypton was the closest to xenon, but there have been problems with erosion in testing which has been a major lifetime limiting factor [7]. The step by step process for finding the mass and size estimates can be found in Appendix B.2.2. Calculations for xenon, krypton, argon, neon and hydrogen was carried out by the Engineering Equation Solver (EES) program. Equation sheets and result tables can be found in App. B.2.3.

The current density is a vital part of the mass and radius calculations. With the assumption that we will be able to have 2 MV/m (where the length refers to the distance over which the voltage is applied)[10], and following the Child Langmuir Law, we were able to come up with a current density of 131 A/m2 for xenon. Calculations concerning the current density can be found in App. B.2.2 (refer to step ③ in size calculations). The assumption of 2 MV/m comes from an extrapolation to future accomplishments. 1 MV/m is already possible. We see this by comparing cesium and xenon in similar conditions with the 1 MV/m restriction. Current densities, found using information from the graph [11], are very similar, meaning that 1 MV/m was a viable assumption. Calculations for this comparison and the graph used are found in App. B.2.4. Comparing cesium and xenon is acceptable because of their similarities in mass (cesium = 133 amu and xenon = 131 amu). We are predicting that in the future, when this ship will be built, we will be able to have 2 MV/m without breakdown. Current densities for the other inert gases were found using the same gap width as with xenon. A table summarizing the current densities, mass estimates, and size estimates for nine thrusters can be found in App. B.2.5.

The mass flow rate needed for each thruster, along with the mass of the propellant, is the same for all of the propellants. The storage for the propellant, though, does increase as the propellants gets lighter because its density goes down, therefore requiring more space and mass to contain them. Calculations concerning the propellant mass and its volume can be found in App. B.2.6.

Propellant Storage

The operational pressure required for xenon propellant is 1e-3 Pa.[6] With this low pressure the volume needed to contain the propellant is large, requiring massive propellant tanks. We will instead store the propellant in a temperature and pressure regulated tank. The xenon will be stored as a liquid at 101.3 kPa of pressure and 161K in temperature with a density of 1987 kg/m^3 [5]. With a well-insulated tank and a polished surface, this temperature will be possible to maintain. Mass flow from the tanks will be controlled by valve systems and released to a heater to be vaporized for use in the thrusters. The storage tanks will be 2024 aluminum with thickness of 2 cm. This thickness was dominated by the requirements for meteor impact resistance and not the pressure in the tank. Calculations determining the thickness and mass can be found in Appendix B.2.7.

Thruster Style and Special Features

The ion source for the thrusters will be electron bombardment. The style will be spherically convergent, with a multiple button grid. Our thrusters will be able to convert electricity to thrust at 85% efficiency [10]. Nine thrusters will be used for our configuration. If something were to malfunction in one of the thrusters it would not have as great of an impact on our ability to get to Mars as if we only had three of four thrusters. The increase in mass from one thruster to nine thruster is approximately three tons. This increase was considered to be allowable to better insure the success of our overall mission.

Our thrusters will also be able to use electric fields to guide the ship. There will be enough deflection to turn our ship around for the deceleration stage of our trip.

A summary list of specifications for the thruster system can be found in App. B.2.8.

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§ 1.3 TRAJECTORY

The trajectory from Earth to Mars, like any planet-to-planet trajectory, will involve three steps. Step one is spiraling out and escaping the initial planet's gravitational field. Step two is the transfer along the trajectory to the target planet. Step three is spiraling in to the target planet along a safe path.

According to NASA, a human being should not be without the effect of gravity for more than a month. As a team we decided not to incorporate artificial gravity into the design, so therefore the maximum time for the three steps discussed above is a month.

In order to make the trip in a month's time, our spacecraft will use low thrust, with a high exhaust velocity. Spacecrafts of this type are good for longer missions, making further, faster missions with higher payloads possible. In order to make our mission practical we have assumed that our thrust producing system, with mass MW, will have to achieve a technical perfection of alpha = 20 kW/kg [2]. The thrust producing system includes the power source, power conversion plant, thrust chambers, structures, controls for thrust producing system, tanks, pipes, valves, etc.

Trip to Mars

Spiraling out from Earth

Since the spacecraft will be quite massive and have low thrust, it must be built and maintained at a station in an orbit about the Earth. This station will establish the rocket's initial conditions. At this "maintenance station," we will prepare for each mission, repair the spacecraft, load and unload passengers, replenish supplies (subsistence items, fuel, propellant), and dispose of waste.

There are five points of equilibrium between the Earth and the moon known as Lagrange points. The second equilibrium, L2, was chosen for the station since it is very stable. It is also an appropriate point because it is the furthest from the Earth (probably the easiest from which to escape), and it is close to the moon's surface (location of resources, i.e. ^3He) [3] . Since L2 is directly behind the moon as viewed from the Earth a relay would probably be needed for communications.

From charts and graphs, a maximum spiraling out time of 1.5 days was assumed [2]. We will use a tangential thrust, which will give the minimum spiraling out time.

Spiraling into Mars

We decided the final conditions will be determined by the same orbit as the inner moon of Mars, Phobos. This orbit is about as close as the craft can get to Mars and still maintain an orbit while avoiding the atmosphere. The orbit of Phobos provides the shortest distance possible for transport of passengers to the surface. In addition, this orbit is almost a perfect circle resulting in a constant distance from the surface of Mars. If the rocket shares an orbit with Phobos, a constant distance between the two will be maintained, facilitating mining of Phobos for more resources.

From charts and graphs a maximum spiraling in time of 1.5 days was assumed [2]. For our purposes a tangential thrust direction will be best.

Transfer from Earth to Mars

There are many different possible transfers from Earth to Mars, each of which can be broken down into three phases.

Phase one will be the acceleration of the rocket up to its maximum velocity, VM, from the initial velocity of the transfer. It obtains this VM in a thrust time T1, travels a distance S1, and requires a propellant mass M1.

During phase two the rocket will not thrust; it will glide at VM for a time T2 and travel a distance S2. The rocket will also turn 180 degrees so that the thrusters are pointing directly opposite to the direction of flight. Assuming only inertia resistance, a transverse thrust can be applied to turn the ship around. The rotation will have to take place in a time less than T2. A thrust of equal magnitude in the opposite direction will stop the rotation. A computer will control this and keep the final orientation locked.

Phase three will be the deceleration of the rocket to the spiraling velocity necessary. During phase three the rocket will travel a distance S3, require a propellant mass M3 and occur in a time T3.

The procedure for determining the optimum transfer for our mission is discussed in Appendix B.3.1-3. We assumed that the exhaust velocity, V, will be the same for all thrust phases. The system variables we decided upon for our mission are summarized in the following table.

Parameter	Value	Parameter	Value
V	170 km/s	S1	2.44e+10 m
VM	90 km/s	S2	5.39e+10 m
M1	4.42e+5 kg	S3	1.91e+10 m
M3	3.36e+5 kg	ST	9.74e+10 m
MW	3.70e+5 kg	T1	8.64e+5 s
PF	0.228	T2	8.98e+5 s
		T3	6.57e+5 s

During phases one and three, a computer will be able to control the trajectory by changing the direction of the thrust.

A schematic showing the trajectory is shown in App. B.3.4.

Replenishing Propellant at the Phobos Orbit

In order to increase our payload fraction (the payload divided by the total initial mass) and decrease the size of the thrust producing system, our spacecraft will only carry enough propellant to get it to Mars. We must replenish our propellant supply for the journey home.

Two ideas emerged as a way to accomplish this. First, we could send the propellant from earth via another spacecraft along the most efficient trajectory. It would not matter how long this trip takes as long as the propellant was in a Phobos orbit for rendezvous with our ship when necessary. Something similar to a Hollman transfer, a two thrust elliptical transfer, comes to mind for this propellant courier ship.

The Trip Home

The trip home would again involve three steps; spiraling out from Mars, the transfer to Earth, and spiraling into L2. We would assume the same initial system for the transfer to Earth as for the transfer to Mars.

Intercepting Mars

The mission must leave L2 only when the orientation of Mars relative to the Earth is such that our spacecraft intercepts Mars. Since we don't know the exact path of our spacecraft, the initial orientations can not be predicted precisely. In 30 days Mars travels 6.22e+10 meters and swings an arc of 15.6 degrees about the sun. So when we begin the trip, Mars must be 15.6 degrees behind the final position of the rocket (App. B.3.4).

Intercepting Earth

The trip home must begin when the orientation of the two planets is such that our spacecraft intercepts Earth. The time it takes for this orientation to occur, from the time our spacecraft enters the Phobos orbit, determines how long the craft is at Mars. In 30 days, Earth travels a distance of 7.78e+10 meters and swings an arc of 29.7 degrees about the sun. Thus, Earth must be 29.7 degrees behind the final position of the rocket when we begin the trip home(App. B.3.4).

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Chapter 2
Space Environment and Material Selection

§ 2.1 SPACE RADIATION

Several environmental stresses are imposed on a spacecraft that is traveling on an interplanetary path. These stresses include the following:

- Cosmic Rays (CR)
- Solar Radiation
- Solar Wind
- Magnetic Fields.

Of these, the last two have little impact on the spacecraft; they are weak compared to the first two stresses (Appendix C.1.1). Therefore, it is crucial to determine the effects of cosmic rays and solar radiation on a space craft.

Cosmic Rays

Cosmic rays consist of high-energy nuclear particles, electrons and photons. Because of their extremely high energy, they are capable of penetrating a thick metal wall, dislocating atoms and breaking the atomic structure of a material.

The two major sources of cosmic rays are Galactic Cosmic Rays (GCR) and Solar Cosmic Rays (SCR). GCR fluxes have extremely high energy and come from all directions in space, resulting in a continuous radiation dose to astronauts and spacecraft. SCR are an extensive source of high energy protons that are produced when a solar flare event occurs. Statistically, there are one to two solar flares a month. Both SCR and GCR are affected by the 11-year solar cycle as follows.

- SCR elements increase as solar activities increase.
- GCR decrease as SCR elements increase.

Contribution of Cosmic Rays to Radiation Doses

The radiation dose to the human body, especially Blood Forming Organ (BFO) doses (5 cm into the body), due to cosmic rays is a primary concern when designing a manned spacecraft.

The main contribution to the radiation dose comes from two sources: protons and neutrons. These incident particles cause direct (or primary) radiation damage to materials (App. C.1.2). As a result of the attenuation (or damaging) processes, primary particles produce indirect (or secondary) radiation, increasing the flux of particles with smaller energies than the primary particles in the material. At the inner surface of the craft's body, for example, the flux of the secondary radiation may be many times greater than that of the primary radiation.

The electronic components, which are the most vulnerable parts of the spacecraft's structure, can tolerate more radiation than humans (App. C.1.3, Tables1-4). However, there is a possibility that a computer may report erroneous results due to the energy flow from cosmic ray elements to the computer's memory chips and/or transistors. In this case, the use of advanced algorithms can prevent such erroneous behavior.

The cosmic rays make the following contributions to the radiation dose:

- GCR contribution to an unshielded astronaut is approximately 60 Rem/year or 4.9 Rem/30 days.
- When a solar flare occurs, a spacecraft near the Earth's orbit about the sun usually receives solar cosmic ray particles continuously, lasting no more than two days on average. Occasionally, a huge flare event occurs, ejecting more energetic SCR. As an example, the BFO dose that would have been received by an unshielded astronaut incurred by the large solar flare of August 1972 was 411 Rem [14].

Because the survivable emergency exposure dose that a person can receive within one day is 100 Rem [5], radiation inside the spacecraft due to cosmic rays, especially due to SCR, must be reduced to an acceptable level.

Detection of a Solar Flare Event

Because solar flares do not occur periodically, prediction of the exact time and day of an event is impossible. However, a reliable method to alert crew and passengers to the occurrence of a solar flare is the detection of a rise in intensity in the region of X rays and gamma rays. A peak intensity of dangerous SCR particles occurs several hours after the solar flare has been detected from the intensity rise of the X rays and gamma rays (App. C.1.4), so there will be ample time to prepare.

Material Selection Criteria for Radiation Shielding

The most important parameter for radiation shielding is the mass of the material per unit area (or density of the material times thickness). However, a thin but dense layer of material cannot withstand the secondary radiation satisfactorily. This means that the radiation shield must have a certain thickness regardless of the density of the material.

Determination of Outer Material Thickness Based on Radiation Protection

The UWFR94 design allows for a maximum radiation exposure to the passengers, other than the trained astronauts, of 5 Rem or less for a 30-day mission. This limit corresponds to the maximum annual radiation exposure level for a radiation worker and was based on the following justifications.

1. Due to the cosmic rays, no reasonable shield thickness will achieve the maximum radiation exposure limit of 0.5 Rem, which is normal for a person on Earth.
2. The exposure limit for an astronaut, 25 Rem/30 day, is too high for normal people. In a worst case scenario, this level of radiation could cause death in some individuals.

3. The exposure limit for radiation workers is a result of extensive scientific research.
4. Because the human body is more tolerant of acute radiation doses, an annual dose can be replaced by a 30-day dose.

The assumptions made for this decision are the following.

- Passengers will not receive significant additional radiation doses after they reach their destination.
- There will be no major solar flares comparable to the February 1956 or November 1960 events, which produced the most penetrative radiation ever recorded.

Radiation Shield Thickness

The radiation shielding must protect the passengers from the galactic and solar cosmic rays such that the maximum radiation dose level, 5 Rem/30 days, is not exceeded.

To achieve this requirement, and assuming that at most one large or two average solar flare events occur in one month, the shield must have a thickness of 6 cm of aluminum for the standard wall, and 15 cm thick (additional 9 cm) for the solar flare safe room. (App. C.1.5). Considering the additional shielding effect of much of the equipment on board, a radiation shield of this thickness should reduce the radiation dose due to a large solar flare to below 5 Rem. For protection against galactic cosmic rays, 6 cm of aluminum allows a radiation level of 3 Rem/30 days.

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§ 2.2 METEOROID IMPACT

Hypervelocity impact with space debris and meteoroids is a design issue in many space applications. The UWFR94 design will neglect space debris under the assumption that orbital altitude will be significantly greater than 1100 km, where debris flux begins to decrease rapidly [1, pp. 79-80]. The following is a discussion of the current meteoroid environment model, the mechanics of hypervelocity impact, and the resulting implications for the UWFR94 design. A rough calculation of penetration depth into the exterior surface shows that the thickness required for radiation protection will amply account for the meteoroid hazard.

Meteoroid Environment Model

NASA's meteoroid environment model approximates meteoroid flux data (the particles of a given mass or greater per square meter per second) from a number of sources as a logarithmic function of particle mass [2]:

(1) $\log_{10} F = -14.339 - 1.584 \log_{10} m - 0.063(\log_{10} m)^2$

for $10^{-12}\,\text{g} < m < 10^{-6}$ g

(2) $\log_{10} F = -14.37 - 1.213 \log_{10} m$

for $10^{-6}\,\text{g} < m < 1$ g

where F is the meteoroid flux, and m is the particle mass in grams. This relation is shown graphically in Fig. 1 in Appendix C.2.1 [1, p. 75].

The relation above is valid for near-Earth orbits; a *defocusing factor* is used to account for the decrease of gravitational influence in higher orbits and deep space. Figure 2 (App. p. C.2.1) [1, p. 76] shows the defocusing factor, which multiplies Eqs. 1 and 2. Note that a vehicle stationed at L2 (about 350,000 km above geosynchronous orbit, GEO) has a defocusing factor $G <$ 0.6.

The probability P of impact by particles of a given mass or larger may be approximated as

(3) $P = GFAt$ x 100% [3]

UWFR94, summarized in Table 1, assumes a thirty year mission and a projected surface area of 288 m^2. Calculations may be found in App. C.2.2 . It is seen that the impact threat should not be ignored for particle masses of 1 gram or smaller.

Particle Size (g)	10^{-3}	1	10^2
Impact Probability (%)	100	0.069	0.0026

Table 1. Probability of meteoroid impact of given mass or greater

Mechanics Of Hypervelocity Impact

At hypervelocities (speeds from 1- 250 km/s) an impact between two metals is essentially a fluid phenomenon [4]. Both vaporize–the projectile disintegrates if it is sufficiently small compared to the target (as is the case for the UWFR94), while the impacted region vaporizes locally and momentarily flows as an inviscid fluid. This results in the creation of a hemispherical crater. For composites as well, however, instead of a well-defined crater, extensive delamination and peeling occurs due to the highly directional properties of laminated materials.

The extent of damage, as measured by penetration depth, is a function of impact velocity, target strength and ductility, and projectile diameter. Experimental data, as shown in Fig. 3 (App. C.2.3) [6], indicates that the penetration depth to projectile diameter ratio P/d is a logarithmic function of impact velocity, and that crater dimensions scale linearly with projectile diameter. For composites, the damaged surface area may be ten times the initial impact area because of peeling [5]. Damage decreases with increasing target strength, but it counter-intuitively increases with ductility. It seems that more ductile materials take longer to "freeze" back to the solid state, allowing more time for the crater to form [6]. This is good news for composites, which are generally somewhat brittle materials.

Implications For Design

The fact that meteoroid impacts occur at such high speeds has led to the development of space bumpers, or impact shields [7]. At these velocities impact can cause spallation of the impacted surface, in which potentially lethal secondary particles are generated off the inside surface. Space bumpers are generally thin metal sheets, separated from the vessel wall by a small distance, that are designed to prevent spallation by vaporizing at the point of impact. Unfortunately, the intricacies of designing for this phenomenon are complex enough that application of this concept has only been justified to date for extreme environments like The Giotto (European Space Agency) probe's flyby through the dust cloud of Halley's Comet [1].

Griffin and French state [1] that at 20 km/s "a rule of thumb is that a particle of 1 μg will just penetrate a 0.5-mm-thick sheet of aluminum." The proper selection of exterior materials can often account for the meteoroid risk with a relatively inexpensive mass penalty. An example is the case of the Viking Orbiter propulsion system, where extra precautions were deemed necessary for the fuel tanks because of the deleterious effects of stress concentrations in pressure vessels– stress concentrations that could be caused by non-penetrating impact of particles on the order of 1 μg. Protection was simply incorporated into the outer layer of the tanks' thermal blankets with the addition of lightweight Teflon-impregnated glass cloth [1]. The general response of different materials to hypervelocity impact is discussed in detail in Ref. [6].

Calculation Of Penetration Depth

Unfortunately, both theory and experimental data are nonexistent for impact velocities greater than 50 km/s, while the UWFR94 will be traveling at speeds of up to 200 km/s. Therefore it is crucial that extensive testing be conducted before the final design is implemented. Still, all research to date indicates that the thickness required for radiation protection (6 cm of aluminum) will amply account for the meteoroid hazard.

A number of assumptions had to be made to arrive at a rough upper bound for penetration depth. The p/d ratio was extrapolated to 200 km/s from data gathered at velocities around 10 km/s. Projectile mass, here a function of diameter, was assumed to be the only relevant projectile material property. Penetration depth, not surface area damage, was chosen as the driving criterion for impact protection, since the probability of two meteoroids striking the same region before repairs could be made is unlikely. Therefore, from Fig. 3,

$$p/d \leq 20 \tag{4}$$

and with a maximum meteoroid mass of 1 gram (from Table 1), assumed to be iron, of diameter $d = 0.634 \text{ mm}$ ($1/40$ in.), the maximum penetration depth is

$$\boxed{p = 1.268 \text{ cm.}}$$

Calculations may be found on App. C.2.4. It is seen that the thickness required for radiation protection is much larger than that needed to prevent perforation. Hence, meteoroid protection can be achieved without the complexity and mass penalty of an impact shield.

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§ 2.3 MATERIAL SELECTION

Material selection for our ship involved choosing an appropriate material to use for the outer surface of the living modules. Several materials were considered, and Aluminum-2024 was chosen to be used with a 6 cm standard wall thickness and a 15 cm thickness for the solar flare safe room.

Selection Criteria

The following properties were used as the selection criteria for the material to be used on the outer surface of our fusion rocket:

1. Low mass
2. High strength
3. Fracture and fatigue resistance
4. High elastic modulus (stiffness)
5. Corrosion resistance (sublimation, erosion)
6. Controlled thermal expansion
7. Radiation tolerance
8. Ease of manufacture
9. Ease of modification

In choosing a suitable material both metals and advanced composites were considered. Based on the above selection criteria an aluminum or titanium alloy would best meet the criteria for metals, and metal matrix composites are the most suitable composite.

Metals

Metals provide a homogeneous and isotropic material which pose no problems due to sublimation. Susceptibility to corrosion and cracking can be limited if the metals are treated. Both titanium and aluminum provide reasonably low mass and sufficient strength, and in general are appropriate metals for use in space structures given the selection criteria. Titanium has the higher strength, but also a higher density. Most importantly however, titanium and its alloys display reduced strength at low temperatures, which make the material prone to buckling. In comparison, aluminum has a lower strength than titanium, but it offers many advantages including superior low temperature behavior. Aluminum alloys have greater tensile strength at sub-zero temperatures than at room temperatures. Aluminum is also a cheaper and more easily manufactured material.

Factor of Safety

If metals such as aluminum or titanium are used, the factor of safety for spacecraft is typically 1.4. This provides 95% confidence reliability that exceeds allowables by a comfortable margin [4].

Advanced Composites

Advanced composites can be divided into the three main categories: polymer, ceramic and metal matrix material. Refer to Appendix C.3.1, for a complete comparison and summary of the different matrix materials [2]. The matrix generally determines the physical properties, such as chemical resistance, and it also controls the manufacturability. The mechanical properties, such as tensile strength and elastic modulus, are determined by the reinforcement used in the composite.

Although they are the most massive of the composite matrices, metal matrix composites (MMC) best meet the selection criteria. Specifically, titanium MMC is most suitable for our application given the selection criteria. Please refer to App. C.3.2 for complete description of titanium MMC. The advantages of metal matrix composites include high strength to mass ratios and radiation tolerance comparable to regular metals. However, metal matrix composites present many disadvantages including high variability of material properties due to differences in manufacturing, high cost, and general uncertainty in behavior due to the relatively new and undeveloped methods of testing.

Factor of Safety

The factor of safety required for composites is higher than that for metals due to the fact that testing and behavior characteristics are not as well known. Standard factor of safety for composites is 1.5 if the material properties have been well defined in a rigorous manner and design verification has been done by full scale testing to failure. If design verification has not been done to failure, than then a factor of safety of 2 is required, and if material properties are not well defined, a factor of safety of 3 is needed.

Please refer to App. C.3.3 for a summary of the general comparison between metals and composites. In order to select the best possible material, thickness requirements for the walls of the living modules were determined so that a mass comparison for the structure could be made. There are three factors which control the thickness of the cylindrical sections of the spacecraft. These factors are: internal pressure, meteoroid penetration, and radiation protection. A thickness requirement was calculated for each of the factors.

Thickness Requirements

Internal Pressure

Cylindrical pressure problems are either modeled as thin or thick walled. The thin walled assumption can be used if the ratio of inner radius to thickness is greater than 10. This is the case for our cylindrical living modules. Pressure vessels have two stresses generated: hoop (circumferential) and longitudinal (along the length of the cylinder). This stress are formalized by the equations:

$$sh = p * ri/t$$

$$sl = p * ri/2t$$

w/ p= internal pressure (gage)
ri= inner radius
t= thickness

At most the gage pressure will be 101.3kPa (14.7 psia or 1 atm). The cylinders will be approximately 3m (10 ft.) in radius. The material in question must have a higher yield stress than the stresses produced by the internal pressure. It is evident that the driving stress will be the hoop stress since it is twice the magnitude of the longitudinal stress.

By simply equating the yield stress and the hoop stress a minimum thickness can be found. Upon doing a sample calculation with titanium sy = 825 Mpa (120ksi) a minimum thickness was found to be 3.11 e-3 cm (1.22 e-3 in). Due to the trivial magnitude of this thickness, it is apparent that the internal pressure will not be the driving the thickness of the cylinders. In addition, the low thickness supports the previous assumption of a thin wall pressure vessel.

Meteoroid Penetration

A thickness of 1.268 cm is required to prevent perforation of the outer surface of the living modules by a meteoroid with mass of 1 mg and diameter of .634 mm assumed to be made of iron. Please refer to page 53 of section 2.2 for complete explanation of this calculation.

Radiation Protection

The thickness required for radiation protection is a function of the areal density. Please refer to page 47 of section 2.1 for complete explanation of the thickness required for aluminum and titanium. Because the density of the titanium metal matrix composite is very close to the density regular titanium, the same thickness is required for these two materials.

	ALUMINUM	TITANIUM (metal and MMC)
Density	2770 kg/m^3	4400 kg/m^3
Thickness required		
standard wall	6 cm	4 cm
safe room	15 cm	10 cm

Results

Knowing the thickness requirements allowed us to eliminate the titanium metal matrix composites as a material possibility. Seeing that such great thickness is required, both aluminum alloys and titanium alloys provide sufficient strength without the complications of variability, manufacturing and other disadvantages previously discussed. Therefore calculations (refer to App. C.3.4) of approximate mass were made for both the aluminum and titanium giving the following results:

	Aluminum alloy	Titanium alloy
Mass of living modules:	274, 000 kg	286, 000 kg

Given the lower mass of the ship using aluminum and the advantages that this material offered in low temperature behavior, ease of manufacturing, cost etc., (refer to App. C.3.5 for complete listing of advantages) the team decided aluminum was the best choice for the outer living module material.

Aluminum alloy 2024, a commonly used alloy for space application, was chosen as best meeting the selection criteria. Complete specifications for this alloy are listed in App. C.3.6.

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Chapter 3
Operations Support

§ 3.1 INTERIOR ENVIRONMENT

Artificial Atmosphere

The basic conditions for interior artificial atmosphere were determined so that the health and safety of the crew would be maintained during the trip. Many physical parameters were considered and consequently an environment of 70% nitrogen - 30% oxygen pressurized to one atmosphere was decided upon.

Total Pressure

The boundaries on the total internal pressure in the living modules are bracketed. The upper limit is 101.3 kPa. The lower limit is bound by the desire to keep the alveolar partial pressure of O_2 high enough to let O_2 intake through the lungs operate normally. The lower limit has been determined experimentally to be 18.6 kPa (2.7 psia) [1]. Thus, there is a relationship between the amount of O_2, PO_2 (partial pressure), and total pressure. There is a desire to keep the cabin pressure close to a minimum since this will lend to thinner walls , and lower structure weight. The minimum structural weight range is from 34.5 kPa to 48.2 kPa (5 to 7 psia) , Appendix D.1.1. This range is also suitable for physiological constraints, App. D.1.2. To capitalize on the ability to reduce structural weight, an internal pressure of 34.5 kPa (5 psia) is the best choice. However, the wall thickness needed to absorb radiation is many times that than needed to contain 101.3 kPa, App. D.1.3. Thus, 101.3 kPa will be used as the internal pressure. This higher pressure also lowers the amount of O_2 in the cabin therefore reducing combustibility problems.

Oxygen

The two crucial factors for oxygen are the percent O_2 and the partial pressure of O_2. Very high percentages are ruled out due to the combustibility and toxicity of O_2. Very low percentages are unacceptable because of human lung intake limitations, hypoxia [1]. According to App. D.1.2 the range of acceptable O_2 percentages at 101.3 kPa is approximately 30%. This would mean the remaining 70% would be comprised of diluent gas and CO_2. The low level of O_2 (30%) reduces fire hazard greatly.

Carbon Dioxide

CO_2 will be constantly produced by the crew at a rate of 1kg/man-day [1]. It is important to keep CO_2 levels less than 4%, since amounts greater lead to sickness, App. D.1.4.

Diluent gas

A diluent gas is needed to combat flammability of O_2. It has been suggested that flammability isn't much of a concern when operating in zero gravity, since there is an absence of natural air convection. If this is indeed the case, one would simply have to turn off the artificial convection fans if a fire started . This does not seem to be a reasonable solution, and research at this time is inconclusive [2]. Through preliminary research the three most promising diluent gases are:

1. Nitrogen (N_2)
2. Helium (He)
3. Neon (Ne)

The pertinent criteria for selecting a diluent gas are:

a. **Inertness**- decrease flammability

Helium appears to be safer than N_2 as far as spark-ignition parameters are concerned, but is much poorer in the case of burning rates of substances. N_2 is also a little safer in the burning rates of fabrics. Neon lies in-between and N_2. It may well be the percent of O_2, not diluent gas, that is of importance when dealing with fire hazard. This is supported by the following quotation, "The choice of inert gas will probably be determined by factors other than combustion parameters" [2].

b. **Physiological problems** - decompression sickness
- voice propagation

Decompression sickness occurs when the ambient pressure changes by a considerable degree. These changes could occur by meteoroid penetration, internal explosion, or an undetected leak. The change in pressure results in the formation of small gas bubbles in the body fluids. The bubbles cause the persons pain and discomfort. The effect the bubbles have upon a person is dependent on the number, size, and position of the bubbles. In addition, the bubbles are influenced by the partial pressures of the gas inside. Therefore decompression sickness severity depends on the diluent gas. The severity is measured by a quantity called the *bubble factor*. There are four basic cases for the formation of bubbles and the bubble factor can be looked at for each case and gas [3].

1) Bubbles forming autochthonously in adipose-
This type of bubble is a factor in the generation of the most serious neurocirculatory collapse syndrome. For this case it has been found that Ne is nearly 4 times as safe as He or N_2 [3].

2) Bubbles forming intravascularly in adipose tissue-
These have the potential to disrupt tissue and produce fat emboli. Obstruction of capillary flow is also possible. It has a two stage formation (early and terminal). In the early stage, Ne has a lower bubble factor than N_2 or He. The same is true for the terminal stage [3].

3) Bubbles forming intravascularly in adipose tissue or muscle-
Formation is very similar to case 2. Again Ne is better than He or N_2 [3].

4) Bubbles forming extravasculary as a gas pocket in connective tissue-

This is the most frequent occurrence in decompression sickness but is the least dangerous from a medical standpoint. Ne once more is the best gas [3].

c. **Audibility**

There is also some concern for the possibility of an "explosive" decompression which results in over distension and disruption of the lungs. One might be able to disregard this since the crew isn't very likely to survive such a catastrophic event. For this scenario another non-dimensional quantity can be analyzed, the *Hazard factor*. Helium (.53) is the least dangerous, followed by Ne(.9) and N2(1) [3]. To conclude, it is evident that Ne is the best diluent gas to combat decompression sickness.

Audibility is essential for the crew to maintain communication. The selected diluent gas will act as the medium for voice initiation and propagation, so it must not interfere with communication. At 101.3 kPa with He (70%) communication is a problem. Significant problems would not be expected in intelligibility of spoken voice in cabins of 50% He and fewer in 30% He, and even fewer in Ne-O mixtures [3 p.104]. So the only real gas of concern is He.

d. **Leakage rates**

For long missions gas leakage through storage tanks is of concern. The leakage rate depends on the size of the hole, internal pressure, and the gas itself. Examining App. D.1.5 at a pressure of 101.3 kPa shows the hierarchy for leakage rates. The fastest being N, followed by Ne and then He [3]. These leak rates are independent of the type of storage (pressure or cryogenic).

d. **Storability**

The ease at which one can store the gas is important. Minimizing volume and weight of storage devices is desirable. Gases may be stored in two ways:

i. High pressure vessels at ambient temperatures
ii. Cryogenic storage (low pressure)

High pressure vessels work quite well for both O2, Ne, and N2. The optimal design parameters are shown in App. D.1.7[4]. Using the optimal design parameters, the weight and volume of the storage tanks are:

Oxygen: Vol=.661 m radius sphere Mass= 831 kg
Nitrogen: Vol= 1.22m radius sphere Mass= 2043 kg

{note: all gases contained in Ti C-120 spherical pressure vessel, with factor of safety 1.88}

These volumes and weights are very acceptable when looking at the magnitude of the entire ship. Exact data on the storage of Ne is scarce however, and it is believed that Ne is similar to He since both behave similarly to ideal gases. Ne has an obvious advantage in that its density is 1/5 of that of He, so its weight penalty and volume penalty will be a fifth that of He[4]. Taking that into account it is very apparent that in pressure storage Ne is at some advantage to N2 and He is very poor.

All of the gases can be stored easily in at least one of the three cryogenic storage methods (supercritical, sub critical, and sub critical 2-phase). He can be ruled out due to its excessive weight penalty (3.8) [3]. A decision has not been made regarding the selection of a storage system.

In conclusion it appears that Ne is easier to store in a pressure vessel and can be stored equally well in a cryogenic system. He is definitely undesirable due to its high weight penalty for both pressure and cryogenic systems. Ne was also superior in decompression sickness. As far as leak rates, Ne comes in second behind N2. Considering inertness, He is the best. As stated before, the inert gas is not the driving force behind combustion. This would seem to indicate that Ne is the best choice for a diluent gas, but, Ne has two drawbacks: it is very expensive and there is little hard data on. Therefore, for practical and economic reasons N2 will be the diluent gas.

Crew Heat Loss

It is desirable to keep the human body at certain temperatures, for health and comfort. Usually a skin temperature of 312.4 K (94F) and an ambient temperate of 299 K (70 F) is sufficient. To acquire these temperatures, a study of heat transfer between a crew member and the ship was completed.

Heat loss from a person can occur by three modes of heat transfer.

1. Radiation (Qr)
2. Convection (Artificially induced since at zero g no natural convection exists, Qc)
3. Evaporation of H20 from lungs and skin (Ql)

-Conduction will be neglected-

The factors which can be controlled by subsystems on the ship that will affect the heat transfer are:

1. Temperature (cooling, heating)
2. Air velocity (forced convection)
3. Humidity (evaporation)

The energy balance between a person and their surroundings, assuming no heat storage is

$$Qp - Qw = Qr + Qc + Ql \quad (Qw\text{:wall}, Qp\text{:person})$$

Analysis of each mode of transfer is as follows

1. Radiation

$$Qr = s * fcw * Ar*(Tc^4 - Tw^4) = (2.65*10^{-8})\ e\ (Tc^4 - Tw^4)$$

Assumptions:

1. Uniform clothing surface temperature
2. Enclosure of greater than 9.3 m^2 (100 ft^2)
3. Fcm (gray body view factor) = e
4. Ar= single crew member's surface area= 1.45 m^2 (`15.6 ft^2) [4 p.9]

The result is: Qr=87.86 W/person/day

2. Forced Convection

First there is a need for a relationship between the clothing temperature and the skin temperature:

$$Tc = Ts - L/k\ [(Qc - Qr)/ Ar]$$

L/k is a function of geometry and clothing material which is usually chosen as one, for conservative reasons.

The equation for the rate of heat transfer is the usual convection equation:

$$Qc = hc\ Ar\ (Tc - Ta)$$

The convection coefficient, hc, is dependent upon not only the velocity and geometry but also upon different atmospheres. For our case, this is an O2-N2 environment. The velocity of air in the ship is 6.1 m/min (20 ft/min). This is the maximum a person can tolerate without drafts and papers being blown. Using App.D.1.8 with these constraints a convection coefficient of .5

is the case. It can also be shown with App. D.1.9 that a velocity of 6.1 m/min (20 ft/min) will give a skin temperature of 312.4 K (94 F) with an ambient temperature of 299 K (70 F). Application of the heat equation gives a value of:

$$Qc = 30.4 \text{ W/person/day} \qquad [4]$$

3. Evaporation

Evaporative heat loss can be obtained from the use of App. D.1.10 with the parameters: V= 6.1 m/min (20 ft/s), O2-N2 environment, P= 101.3kPa (14.7 psia), and dew point temperature of 285.2 K (45 F) (see humidity sec.). This data yields the result:

$$Ql = 366 \text{ W/person/day}$$

To compute the heat balance between a crew member and their surroundings, one needs only to add up these losses. It is required that the total of the losses is within the range of heat a person can generate. Since a person can generate a wide range (60.43 W- 1281W) of heat this is not a very hard condition to meet.

$$\text{Sum} = 484.3 \text{ W/person/day}; \; 60.43 < 484.3 < 1281$$

To conclude, the heat generated by the crew will be a required value when computing the overall heat transfer between the ship and space. These numbers will help when trying to ascertain the amount of heating that will be needed in the ship and if insulation will be required.

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§ 3.2 THE LIFE SUPPORT SYSTEM

To transport 100 people to Mars, we need to design a system which will maintain human life during the journey. In this life support design, we will assume the average human mass is 75 kg/person (= 165.34 lbm/ person). For 100 people, the total mass is 7,500 kg (= 16534 lbm).

This chapter is concerned solely with the requirements of manned space flight. Many factors are considered that are either essential or desirable in maintaining healthy people for the duration of the 30 day journey. The first part of the chapter is concerned with the most basic requirements to maintain life including suitable temperature and pressure in the living modules. The next part of the chapter deals with the food and water supplies, and the disposal of waste products. The final part of the chapter, in less detail, deals with other aspects of living in the new habitat, such as sleeping, exercise and recreational opportunities.

Basics Requirements

To maintain life in space, we need the basics requirements. According to Sharpe[8], the minimum life support system must function to meet the following needs:

Requirements	
Metabolic oxygen	.91 kg/person-day
Drinking water	3.63 kg/person-day
Hygiene water	5.44 kg/person-day
Food	.95 kg/person-day
Waste Production	
Carbon dioxide	1.02 kg/person-day
Water vapor (perspiration and exhaled breath)	2.49 kg/person-day
Waste wash water	5.44 kg/person-day
Urine	1.45 kg/person-day
Feces	.16 kg/person-day
Metabolic heat	12660 kJ/man-day

In our mission, we plan to transport 100 people from Earth to Mars in 30 days. The people will include both adults and children. Here, the calculation of the basics needs for 100 people in 30 days and the metric conversion are listed.

Use : 1 ton = 2000 lbm
1 lbm = 0.4536 kg
1 Btu = 1.055 kJ

Requirements		
Metabolic oxygen	6,000.0 lbm	= 2,721.6 kg
Drinking water	24,000.0 lbm	= 10,886.4 kg
Hygiene water	36,000.0 lbm	= 16,329.6 kg
Food	3,900.0 lbm	= 1,769.04 kg
	Total	= 31,706.64 kg
Waste Production		
Carbon dioxide	6,750.0 lbm	= 3,061.8 kg
Water vapor (perspiration and exhaled breath)	16,500.0 lbm	= 7,484.4 kg
Waste wash water	36,000.0 lbm	= 16,329.6 kg
Urine	9,600.0 lbm	= 4,354.56 kg
Feces	1,050.0 lbm	= 476.28 kg
	Total	= 31,706.64 kg
Metabolic heat	36,000,000 Btu	= 37,980,000 kJ

Atmosphere in space flight

Air pressure inside the orbiter = 1033 grams per square centimeter
= 14.7 pounds per square inch

One of the advantages of using near-atmospheric pressure in the modules is that the stress imposed on the people due to atmosphere re-adjustment is very small.

Air is made up of : - 70 % nitrogen

- 30 % oxygen

Compared with Earth's atmosphere : - 78 % nitrogen

- 21 % oxygen
- 1 % other gas such as argon and neon

Temperature in the orbiter (space flight) is between 16 and 32 degrees Celsius (61 and 90 degrees Fahrenheit).

Supplying oxygen for the cabin is only one part of the overall task of the atmospheric-control system. It must also remove carbon dioxide, solid particles, other contaminants, and water vapor. According to Sharpe[8], the removal of carbon dioxide can be done by chemical, mechanical or physical means. American and Soviet spacecraft all use a chemical system.
There are two ways to supply oxygen, and to remove carbon dioxide, solid particles, other contaminants and water vapor. They are

1. The American method

By using lithium hydroxide (LiOH),

$$2\,LiOH + CO_2 \longrightarrow Li_2CO_3 + H_2O$$

In reality, 1 lb.(= 0.4536 kg) of lithium hydroxide absorbs about 0.8 lb (= 0.3629 kg) of carbon dioxide. The system is relatively simple mechanically, but lithium carbonate (Li_2CO_3) cannot be regenerated, therefore 2.5 lb of lithium hydroxide must be carried for each person-day in the journey. For 100 people and 30 days journey, we need to bring 7500 lb or 3402 kg of lithium hydroxide.

2. The Soviet method

By using superoxide (NaO_2),

$$2\,NaO_2 + CO_2 \longrightarrow Na_2CO_3 + 1.5\,O_2$$

$$2\,Na_2CO_3 + 2\,CO_2 + H_2O \longrightarrow 2\,NaHCO_3 + 1.5\,O_2$$

The advantages of using superoxide are

- removing water vapor
- producing oxygen

Food

Each person needs at least 0.59 kg / day (= 1.3 lb / day) of food and the menu provides 2800 to 3200 calories per person-day.
The food must provide an average caloric distribution of about :

- 17 % protein
- 32 % fat
- 51 % carbohydrates.

The carbohydrates and the fat supply the energy to the body. Protein also provides energy, but it is primarily needed to build up the body tissues or make normal wastage. Finally, various elements such as iron, calcium and phosphorous are required by the system, together with a regular supply of the various vitamins. According to Bourland, Fohey, Rapp and Sauer [2], to maintain good nutrition, the menu should provide at least the following quantities per day :

Protein	(g)	56	Vitamin B	(μg)	3.0
Vitamin A	(iu)	5000	Calcium	(mg)	800
Vitamin D	(iu)	400	Phosphorous	(mg)	800
Vitamin E	(iu)	15	Iodine	(μg)	130
Ascorbic Acid	(mg)	45	Iron	(mg)	18
Folacin	(μg)	400	Magnesium	(mg)	350
Niacin	(mg)	18	Zinc	(mg)	15
Riboflavin	(mg)	1.6	Potassium	(meq)	70
Thiamine	(mg)	1.4	Sodium	(meq)	150
Vitamin B	(mg)	2.0			

An 8 day example menu can be found in Appendix D.2.1.

Difficulties in dealing with the food in zero-g conditions have resulted in food made up into a paste form which can be eaten from squeeze tubes. However, this would not be satisfactory for 30 days journey. Using the experimental results from Apollo's trip to the moon, the food remains stable if it is stored under the following conditions :

Temperature	-5 C to 60 C
Pressure	19.7 psia to 1*10E-8 psia

Relative humidity	30 % to 90 %
Atmosphere	100 % oxygen
Acoustic noise	135 db.,35 to 4800 c
Gravity	compatible with weightlessness for long periods of time

Another way to keep the food fresh is freeze-dehydration. This is a common way to store food in space. The space foods must be lightweight and need no refrigeration. Most of the weight in the food is water. In fact, 9/10 of vegetables and fruits and 4/5 of meat and fish are water. If it is removed, then the weight is greatly reduced. Only replacing the water makes the food edible.

Food to be freeze-dehydrated is cooked and then quickly frozen in a liquid gas, usually nitrogen. It is then cut into individual servings and placed into a vacuum oven. The pressure in the oven is usually 1.5 mm.Hg or lower, and the temperature is slowly raised to 50 C or 60 C. In this way the ice in the frozen food sublimes or passes directly to steam without first turning to a liquid and wetting the food. Water is thus removed from the food without damaging it or changing its chemistry. The food preparation system is designed to require minimal meal preparation times.

Water supplies

Water is also a crucial element in the expedition. In the journey, we need both hot and cold water. The hot water is used for reconstituting the space food, while the cold water is used for drinking and hygiene. In the command module of the Apollo, people used the potable water machine to provide water. This machine is generated by a 1..42 kW fuel cell that combines hydrogen and oxygen. This machine will provide both cold and hot water.

Waste management

In the spaceceraft, we try to minimize waste. However, we still have to provide the waste management for collecting, storing or disposing of liquid and solid human wastes, uneaten food, and miscellaneous trash. For example, feces from people in the rocket are collected in special plastic bags. Before use, a packet of germicide, in a crushable plastic pouch is placed in the bag. After use, the toilet tissue is placed in the bag, which is then folded and sealed. The germicide is released by crushing the pouch. This kills microorganisms that cause decay and odor. Another example is the waste management of urine. Urine is dumped directly overboard, where it first freezes and then sublimes, or turns directly to water vapor. Other wastes accumulating in the cabin consist of leftover food, dentifrice gum, used facial tissues, and wash cloths. The leftover food can be treated by using a germicide to kill bacteria that cause decay. It is then placed in a sealed plastic bag and stored. Other trash is placed in bags and similarly stored during the expedition. After arriving at Mars, we can throw away the trash bags and recycle some of the trash, such as glasses, plates, and cans of soda.

Living Habitat

Sleeping

Maintaining regular schedules of sleep for crew is important. Each sleeping room is designed for four people and is insulated to prevent noise disturbances. The zero-gravity makes it possible to eliminate things that we on the Earth take for granted, for example, we do not need beds. Instead, each crew member will be assigned a small closet-like enclosure equipped with a zippered sleeping bag. People sleep erect against the interior walls. There is an alarm or communication system in the sleeping chamber.

Exercise

The weightlessness can affect health, causing sleepiness, fatigue, motion sickness, hypertension, reduced blood volume and weight loss. To prevent these symptoms, the crew members need to exercise approximately 3 - 4 hours/ day. Exercise examples include use of an ergometer (bicycle), or a simple device consisting of a rubber cord and a handle.

Recreation and communication

Recreation is needed in this habitat to reduce feelings of isolation.
Some examples of recreation are

- books
- games
- tape players
- television and video tapes

By designing the spacecraft to include as many windows as possible, the crew will have the opportunity to exercise distant vision. Communication between people in this small habitat is also important. People can form a small club based on their interests, such as poker club and chess club. They can also organize a small party during the journey so people can gather and socialize.

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Chapter 4
UWFR94 Structure

§ 4.1 DESIGN OF THE STRUCTURE

As a team we evaluated different configuration possibilities for UWFR94. After examining the advantages and disadvantages of the different configurations and features, we drafted the configuration shown in Figure 1 below. Please refer to Appendix E.1 for the complete advantage and disadvantage list for the final UWFR94 structure.

We worked to maintain simplicity while developing a modular structure. We chose to design a modular structure to facilitate ease of assembly and future modification. The structure is comprised of three identical cylindrical living modules with hemispherical end caps. The use of the cylindrical shape provides an easily pressurized vessel.

In order to make the UWFR94 structure symmetric about the center of gravity, we connected the living modules in a triangular or tripod arrangement. The equilateral triangle formed by the living modules provides a stable structure to which the thrusters can be connected. UWFR94 has axial symmetry about an axis through the center of the triangular arrangements.

Once the major geometrical features of UWFR94 were designed, the next step was to determine the dimensions of each of the structural components. We first determined the dimensions of the living modules.

UWFR94 Structural Design

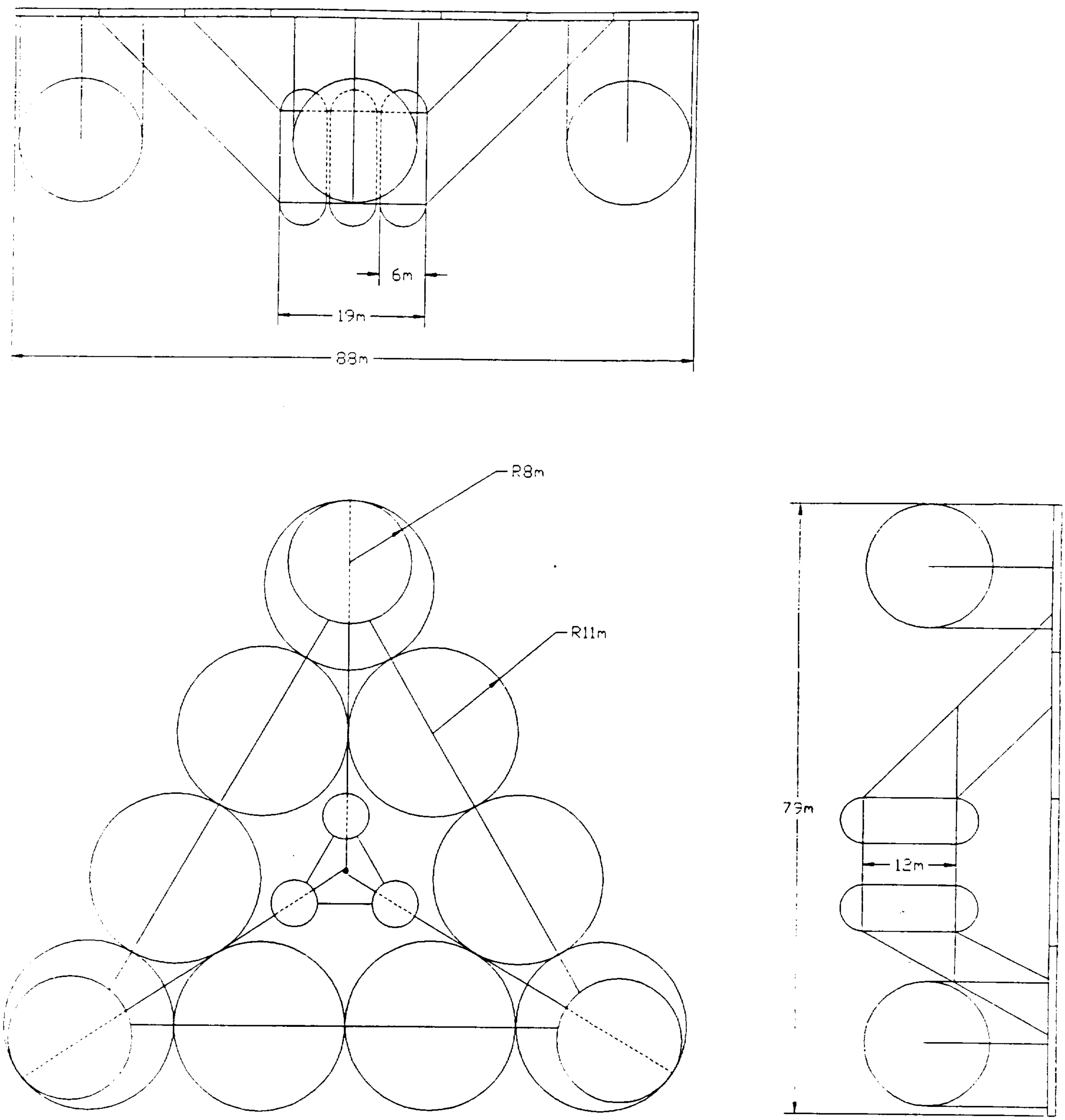


FIGURE 1

§ 4.2 OPTIMUM CYLINDER DIMENSIONS

The dimensions for UWFR94's cylinders, radius r and length l, were calculated for the minimization of exterior surface area, and hence mass, with the method of Lagrange multipliers. The surface area for a cylinder with hemispherical ends was minimized subject to a volume constraint, as described below.

Analysis

The volume of one Fusion Rocket cylinder is

(1) $$V = \pi r^2 l$$

not including the end caps, whose volume $V_{sph} = \frac{4}{3}\pi r^3$ was neglected for two reasons. The physical rationale is that this volume would not be utilized to its full potential. The mathematical justification is that the minimization process results in a cylinder length of zero- a sphere, which has the optimal volume per surface area of all three-dimensional solids.

The surface area for the cylinder and ends is

(2) $$S = 2\pi r l + 4\pi r^2$$

which when multiplied by thickness and mass density gives the overall mass of the cylinder.

The method of Lagrange multipliers may be stated as [1]

(3) $$\nabla f = \lambda \nabla g$$
$$g = 0$$

where ∇ is the gradient operator, f is the function to be minimized (Eqn. 1), and g is the constraint equation (Eqn. 2). It is relatively easy to derive using differential calculus; pedagogically, the Lagrange multiplier λ may be thought of as a variable introduced to enforce a nonzero constraint constant that would otherwise be lost to differentiation.

Solving gives (see Appendix E.2)

(4) $$r = \frac{2V^{1/3}}{(32\pi)^{1/3}} = 0.430V^{1/3}$$
$$l = 4r = 1.720V^{1/3}$$

Thus, from Eqns. 2 and 4, the mass for one cylinder is

(5) $$m = 2.219\pi\rho t V^{2/3}$$

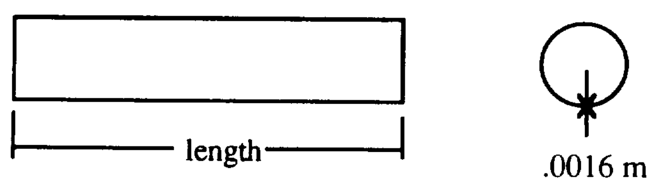
where ρ is the mass density and t is the cylinder thickness.

Note that the optimum $^{l}/_{r}$ ratio was calculated as 4/1 (Eqn. 4). Surface area at a given volume was calculated with EES™ for length-to-radius ratios $^{l}/_{r}$ between 0.01 and 100 (App. E.2), verifying this result [2].

With thirty-three crew members in each living module, the required volume of a single cylinder is approximately 452.4 m^3. To enclose this volume at the optimum $^{l}/_{r}$ ratio, the living modules should be 12 m in length and 3 m in radius.

§ 4.3 THE TRUSS SYSTEM

A truss system is used to attach the modules to the thrusters. We wanted equal support in all different directions, so we chose to have members of circular cross-section. The members will serve only as structural support, not as passageways between modules. To save mass, hollow members were used. Al-2014 [1] was chosen because of its high strength and relatively low mass. A thickness of .16 cm was chosen for the walls of the members. An example a generic member is illustrated below.



There were two factors which we used to determine the location of the members. The first factor was that each member should be placed so the stresses among the members are distributed evenly . The second factor was that each member have one redundancy. There are two forces that apply stresses to the members are acceleration and "people forces". The acceleration force was calculated to be 1100 N, and the maximum people force was calculated to be 4488 N. These calculations can be found in App. E.3.1. The following is a model of our truss system.

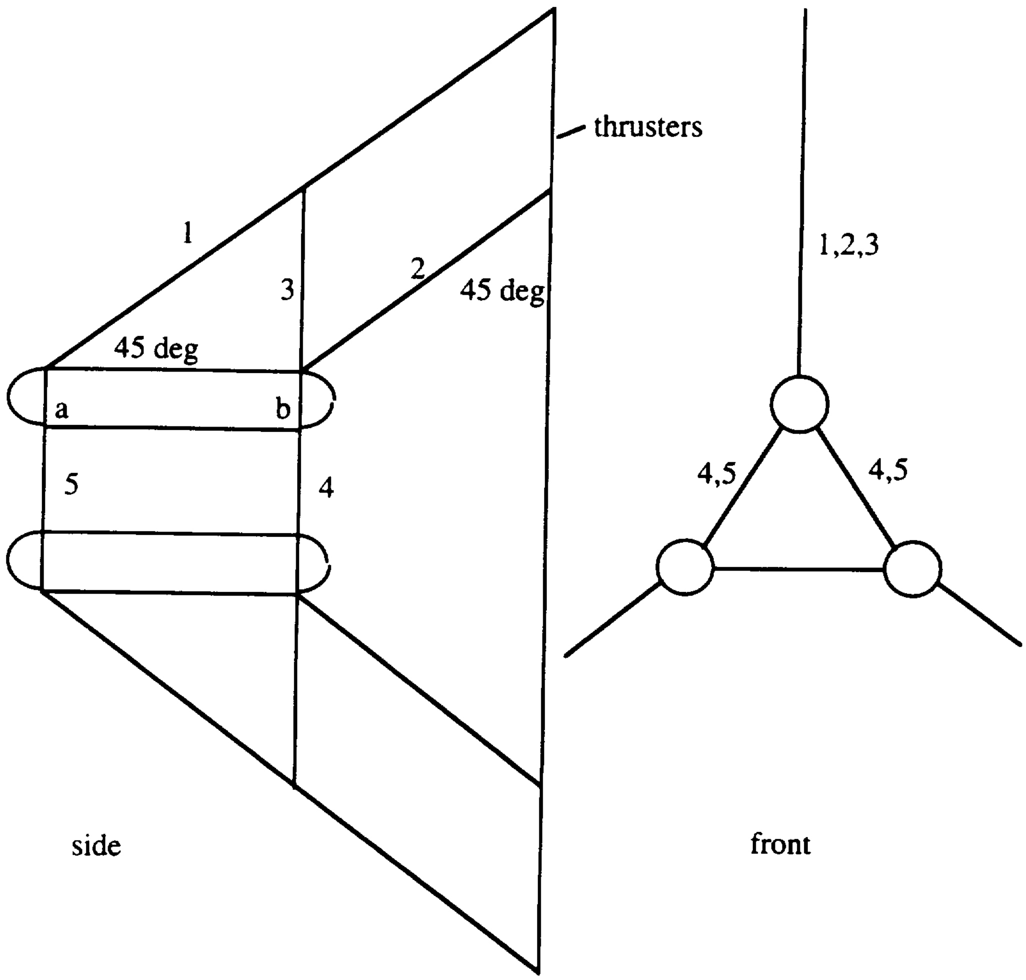


FIGURE 1. The truss system.

A fully dimensioned model can be seen in App. E.3.2. The calculations of the truss system were done in EES and are located in App. E.3.3. Keeping in mind that the thickness of the members is .16 cm, below is a chart of the outer radii and the lengths of the members.

member	outer radii (cm)	length (m)
1	27.9	33.9
2	11.8	17
3	9.2	12
4	8.8	10
5	8.8	10

A safety factor of four was given to each member. Since the total mass of the truss structure is only 1204 kg, there was little penalty in choosing a large factor of safety. The free body diagrams of the members are in App. E.3.4. To achieve these forces, the people force was applied in the direction that brought about the maximum forces on each member.

Placement and Orientation of Each Member

Members one and two are attached to the thruster frame at 45 degree angles, and would therefore provide equal strength in either the horizontal or vertical directions. Forces were calculated in member one as if member two were missing. So if member two somehow became disabled, member one could handle the forces by itself.

What if, for instance, that there is a vertical people force (a force of 4488 N) at position 'a' in the module? Member one could handle this force itself. But suppose there were a vertical people force at position 'b' in the module. If member two were missing, another member is needed. That is why member three is included in the truss system.

Conversely, if member three were missing in the second scenario, member two would carry the weight member three would have.

Suppose member one were missing, and there happened to be a vertical people force at position 'a'. The absence of member one renders member three useless. In this case, members two and five would pick up the forces.

Members four and five connect the three modules to each other. These members are placed at the ends of the modules, to minimize the forces in the members. This is explained in App. E.3.5. Because of the thickness of the modules, there is no danger of them deflecting for fracturing if a people force is exerted at the middle of the module. These members can also be redundancies to other members, and vice versa.

Members four and five were chosen to be ten meters long. We decided that this distance was far enough away that if there were an explosion, for example, in one module, the others would be protected. Ten meters is a short enough length, though, if a spacewalk would be needed for some reason.

This truss system combines redundancy and strategic location. There are many other possible systems that would work just as well.

§ 4.4 ASSEMBLY

The UWFR94 will be assembled in space. One of the most advantageous aspects of the vehicle's modular design is that each of the three living modules will be fully constructed on the ground and completely operational before launch, thus reducing assembly time and overall costs. A Heavy Lift Launch Vehicle, with a payload capacity of 100-150 metric tons, will launch each module to Low Earth Orbit [3]. From there it will be towed to Lagrange Point 2 (L2) by another vehicle, perhaps one similar to the Space Shuttle and based on the moon (also reducing launch costs). The 3 engine/thruster clusters will be launched the same way, although they will be sent up unconnected due to volume constraints [3].

Once the subassemblies have been launched, an astronaut team will rendezvous at L2 with the interconnecting supports. Here, too, the Space Shuttle provides an adequate means of delivering payload and providing a support platform for the team. The supports can be manufactured either on the ground or in space [4].

Bolting has two major advantages over welding for the coupling of the subassemblies. The first is that welding is obviously much more labor-intensive, whether on Earth or in space. If the subassemblies are constructed with attachment plates on Earth, the astronaut can simply "plug and tighten," again at a reduction of time and cost. Second, bolting has a fraction of critical damping of 5-7%, compared to 2% for welding [5]. This increase provides a natural means of better controlling unwanted vibratory motion. Of course, care must be taken with either method to ensure that thermal expansion does not pose a problem.

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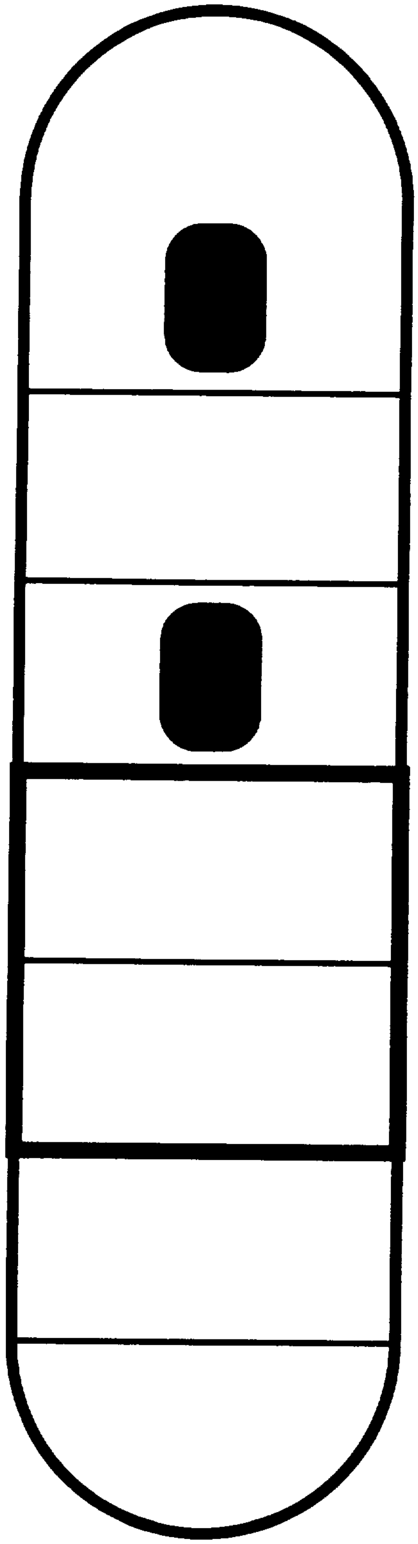
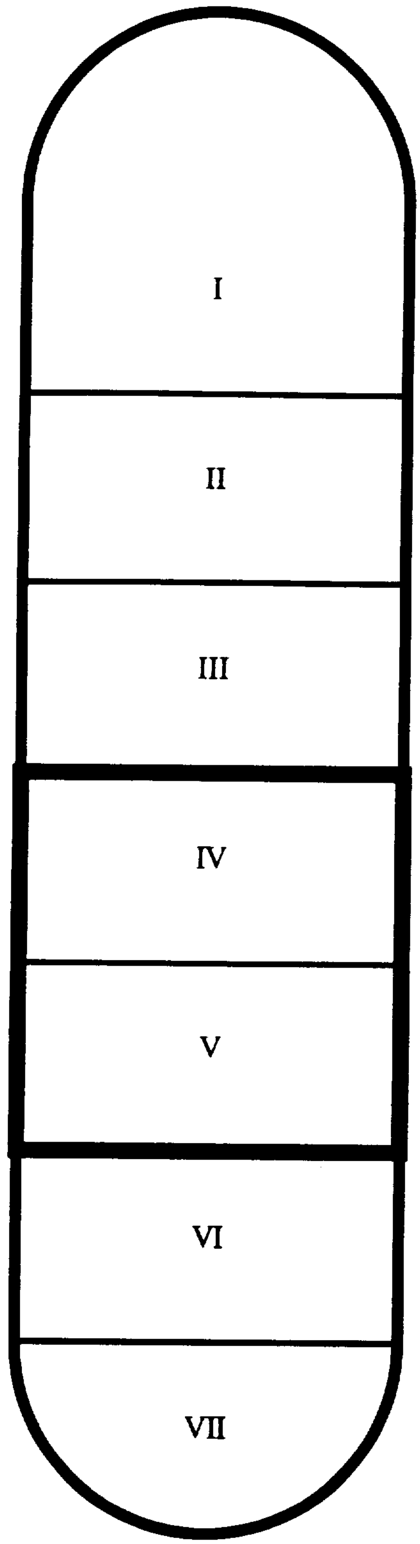
§ 4.5 INTERIOR LAYOUT

PART I



PART II



I. Control unit

II. Sleeping Area

III. Bathroom / Recreation area

IV. Kitchen / Food storage / Dining area

V. Bathroom / Laundry / Storage

VI. Sleeping area

VII. Storage

IV and V are Safe Room.

According to Connors [1], the minimum requirements for humans to live and work in space is 260 ft^3 /person (7.36 m^3 /person) for 1 or 2 months. Thus, for 100 people, the minimum space we need is 26,000 ft^3 (736 m^3). This space determination took into account 4 kinds of functional units

1. work unit (operational task, vehicle management)
2. public unit (dining, recreation, exercise)
3. personal unit (sleeping, personal privacy, personal storage)
4. service unit (toilet, laundry, public storage)

The rocket contains 3 cylinders. Each cylinder is independent, so there are no walking connections between them. As a consequence, each cylinder has the same interior design.

As stated in a previous section 4.2, the dimensions of the cylinders are:

radius = r_i = 3 m
length = l_i = 12 m

The volume of the living area for one cylinder is 365.84 m^3

The total volume of the living area for all three cylinders is 1097.52 m^3

This volume is greater than the minimum requirement of volume per person: 1097.52 m^3 > 736 m^3, thus these dimensions are acceptable.

The cylinder portion of the living area is divided into six floors. One of the caps is also used, so the total is seven parts. They are listed below and pictured in the layout.

- floor 1 : living area for two people / control unit
- floor 2 : sleeping chamber
- floor 3 : bathroom / recreation area
- floor 4 : kitchen / food storage / dining area
- floor 5 : bathroom / laundry / storage
- floor 6 : sleeping chamber
- cap A : storage

The fourth and the fifth floors are the safe room.

The height of each floor is 2 m.

Other dimensions for the thickness are

For the insulation,	$t_i = 6$ cm
For the safety room,	$t_s = 9$ cm
For the standard wall,	$t_r = 6$ cm

The purpose of the insulation is to prevent noise disturbance problems.

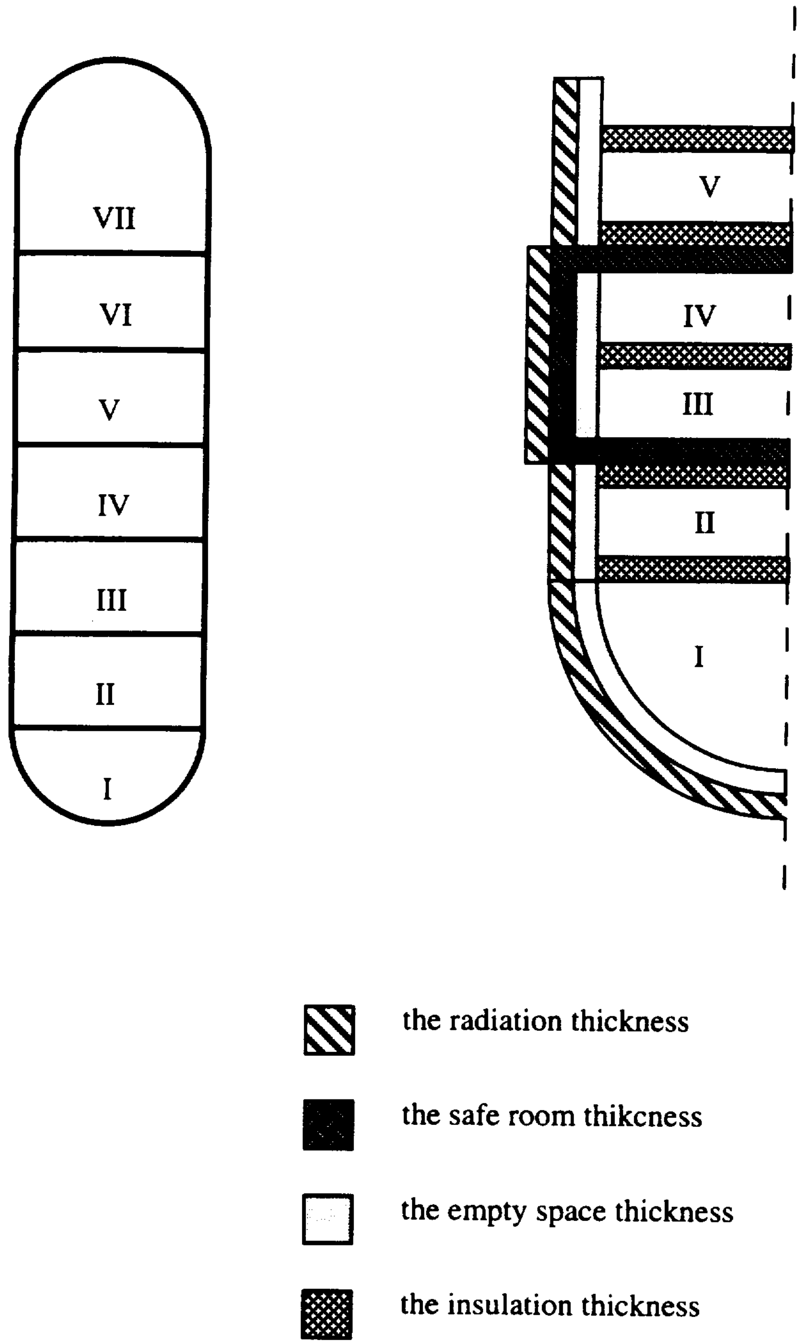
From calculation on App. E.5, option 2 is better than option 1 because

- the empty space in option 2 is larger than the empty space in option 1

  13.5 cm > 9 cm
- there is no empty space at the safe room.

This empty space is used to store the electrical cable and the pipes for water and air circulation.

The dimensions of the option 2 :



The length of outside cylinder (includes insulation thickness and safe room thickness),

$$l_o = 12.54 \text{ m}$$

By ratio of r / l = 1 / 4, the outside radius,

$$r_o = 3.135 \text{ m}$$

For I, II, III, VI, VII, the total outside radius is

$r_{o1} = 3.195$ m

For IV, V, the total outside radius is

$r_{o2} = 3.285$ m

So, the thickness of the empty space for all floors is

$t_e = 0.135$ m = 13.5 cm

From the control unit interior conditions such as temperature, waste management and communication can be regulated. This floor will be divided into three parts. They are the sleeping area, bathroom, and the control unit. On this floor, two people will be stationed to control the above conditions. The sleeping area is a quarter of the total floor area (= 7.07 m^2) and the area of the bathroom is 1/8 of the area of the total floor area (= 3.53 m^2).

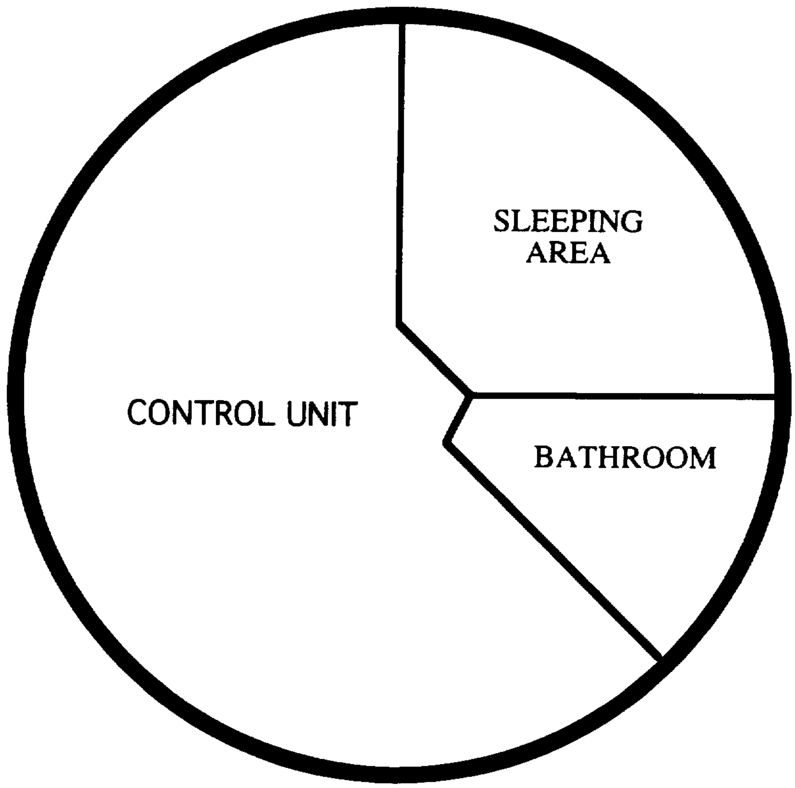
There are two floors designated as sleeping areas. Each floor is divided into four rooms. Each room is available for four people. The area of each rooms is 7.07 m^2.

The bathroom / laundry / storage area is divided into three equal areas. The area of each of these is 9.425 m^2.

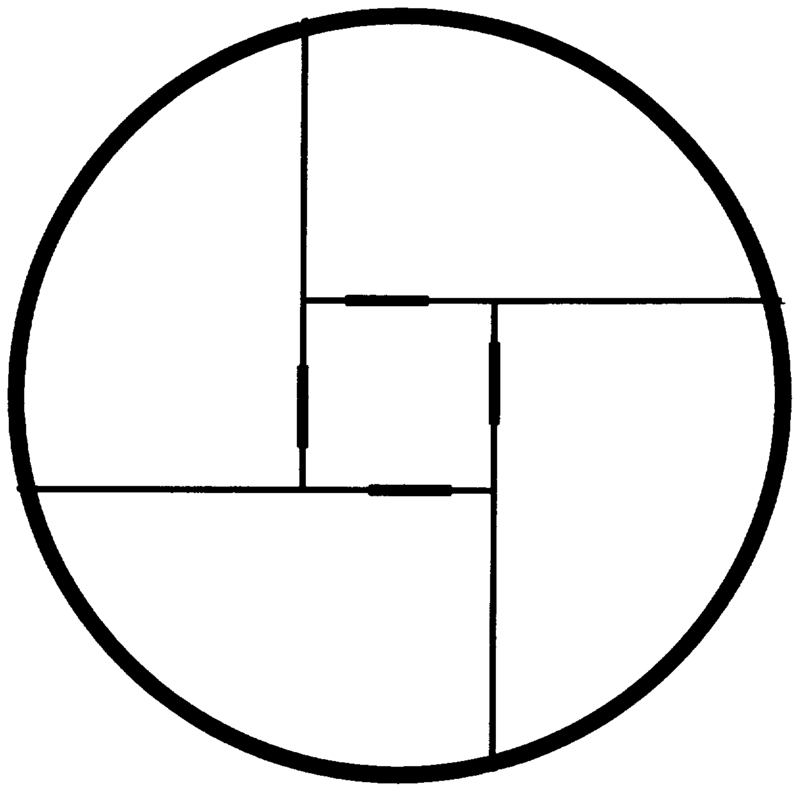
The bathroom / recreation area is divided into two equal areas. The area of each these functional spaces is 14.14 m^2.

The kitchen / food storage / dining area is the place where people can prepare and eat their meal.

CONTROL UNIT



SLEEPING AREA



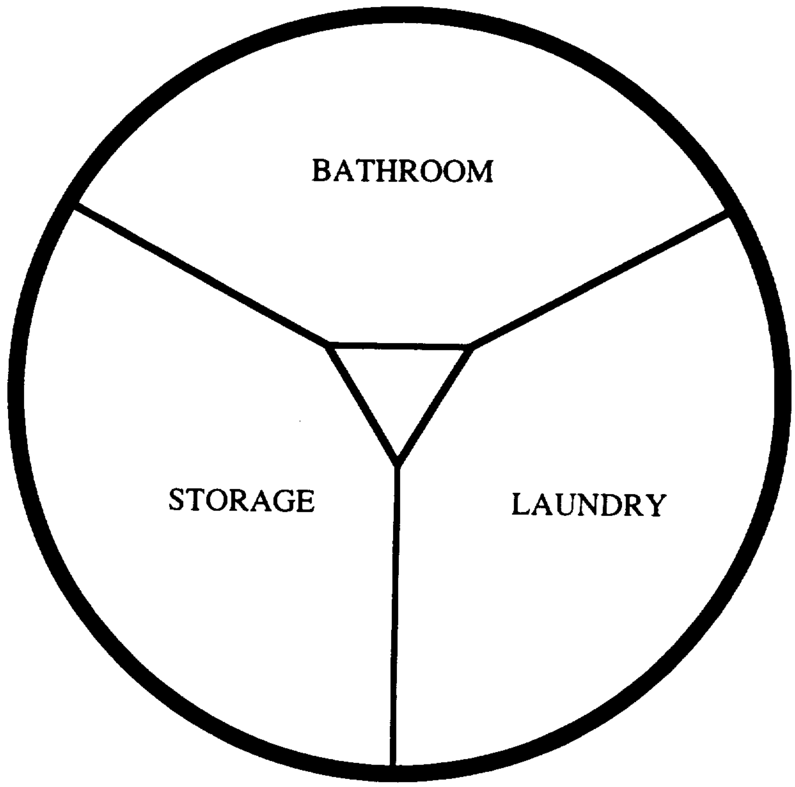
2 sleeping areas : floor II and VI

4 rooms / floor

4 peoples / room

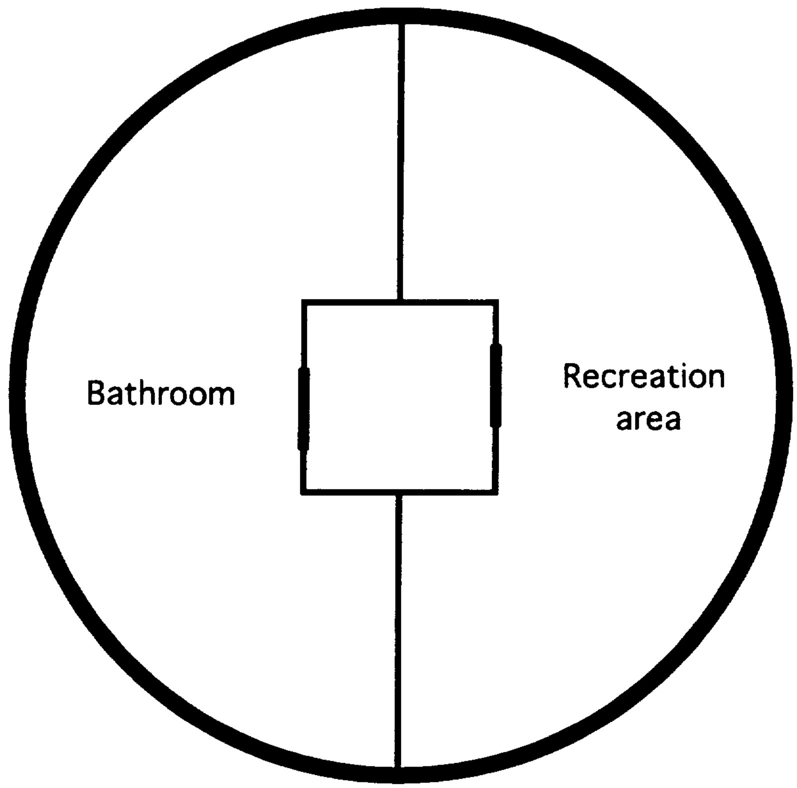
Total = 16 people / floor

BATHROOM / LAUNDRY / STORAGE

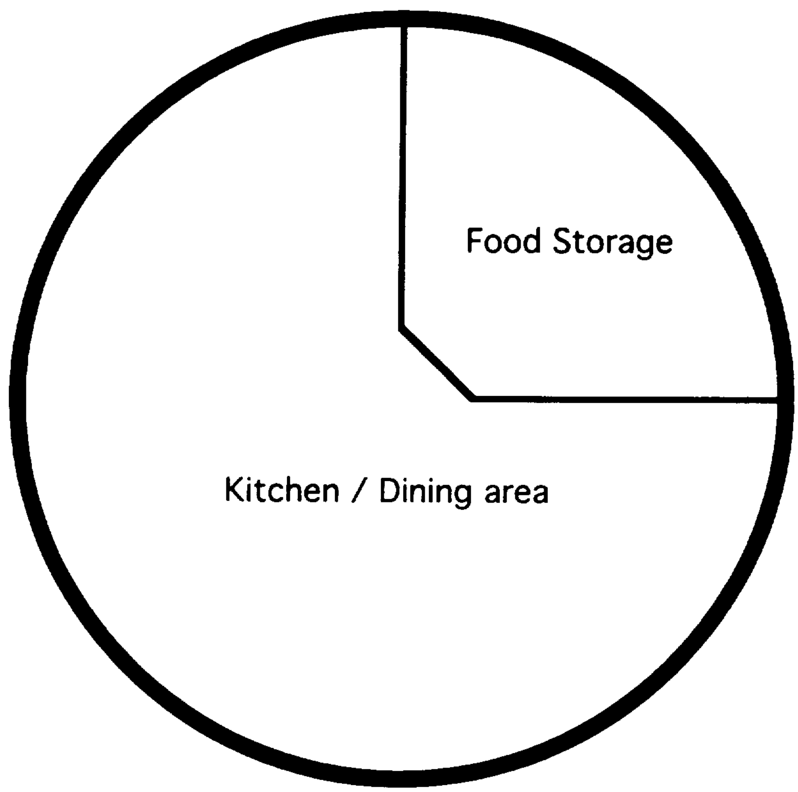


Bathroom, laundry and storage have the same area (= 9.425 m)

BATHROOM / RECREATION AREA



KITCHEN / FOOD STORAGE / DINING AREA



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Recommendations

1. Increase time of travel to reduce size of the propulsion system.

2. Use artificial gravity so that time of travel is not constrained to less than 30 days.

3. Use thruster with multiple exhaust velocities so that there can be different optimum exhaust velocities for each phase of the trip.

4. Investigate possible passageways and/or tranport mechanisms between modules.

5. Work on shuttle connection and landing vehicle.

6. Examine possibility of having windows in ship.

7. Research possible sub-system controls for heat, communication, ventilation, etc.

8. Design system to heat (from liquid to vapor) propellant, and valve system to control vapor in propellant storage system.

9. Do vibrational analysis of structure to find frequency modes.

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