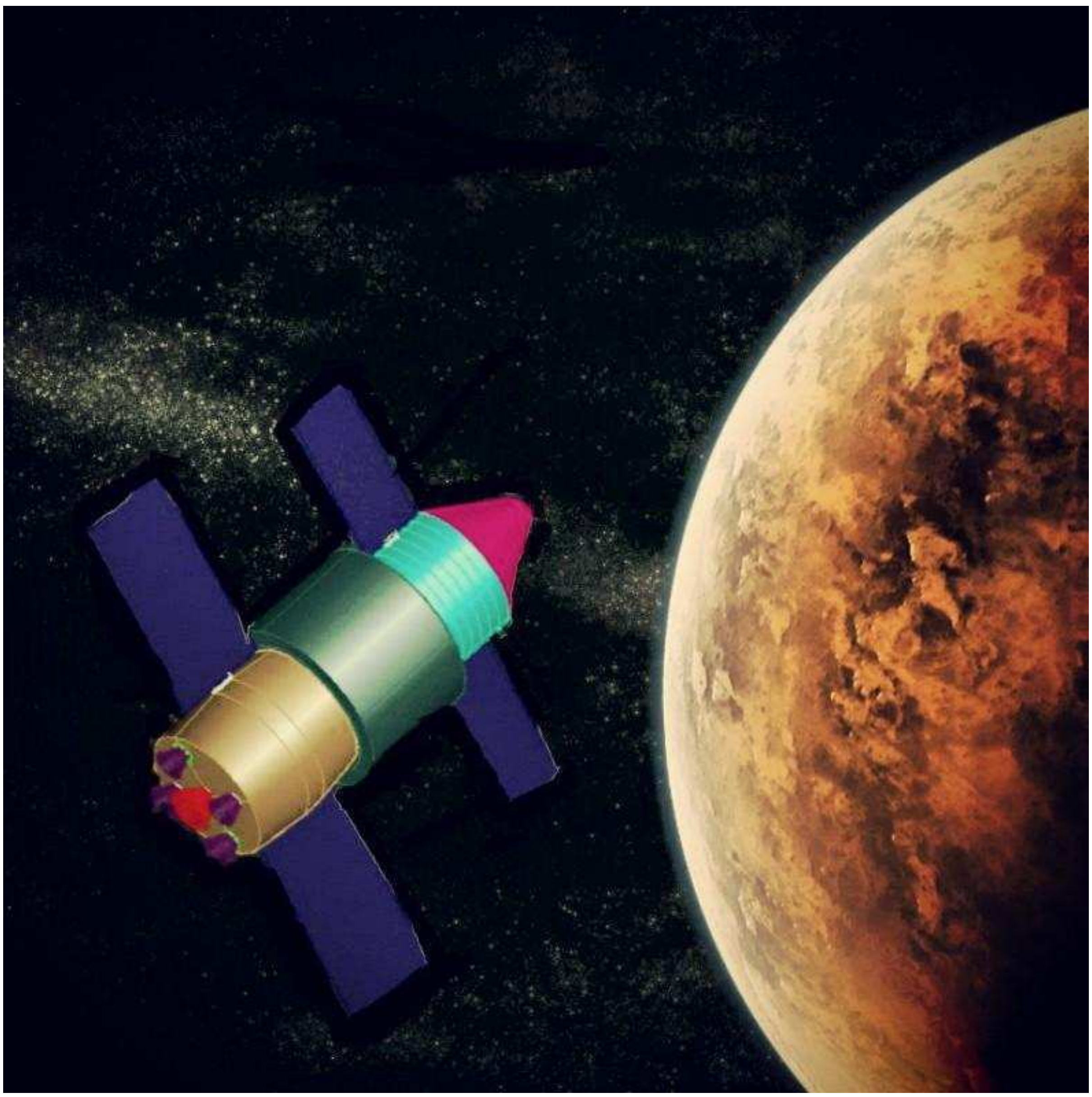
Mars flyby Mission Architecture of Spacecraft
by
Vimana Notion Design Team
India



Vimana Notion Design Team
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***Abstract*— The global space agencies are began the Martian Race to explore the Red planet and searching for the human habitability in other planets. We, the Vimana Notion Design Team proposed the conceptual design of a Spacecraft for Mars flyby mission 2018. The VN Spacecraft has main systems, subsystems, and auxiliary systems for the Mars travel. The main systems are the Power and its distribution systems, Environmental Control and Life Support System (ECLSS), Anti-radiation shield systems. Each main system has subsystems to take care of long duration manned missions, good shielding and auxiliary systems to take care the complete mission requirements like characteristics of liquid hydrogen fuel and water,etc VN Spacecraft is equipped with the habitat module, Crew Transfer module, Service module and propulsion module for accommodations and provisions for crew members of average aged couple and Scientific payloads for the spacecraft. The Anti radiation shield is established with torus-solenoid rings method which is lighter than other methods and the effect of this quadrupole magnetic field on energetic particles is stable in order to shield spacecraft during Mars mission. VN Spacecraft conceptual desugn is also investigated theoretically the Earth reentry conditions and developed some aerocapture options to mitigate G-loads on the returning crew as reported in literatures. It is also describe tradeoffs and studies to develop the Thermal Protection System (TPS) and Systems integration to the launch vehicle. The Angara K7 (Russia) heavy launch vehicle can be used to launch the VN Spacecraft with the weight of 43,000kg approximately to LEO. The safety to the crew members, emergency escape system and all other systems are explained in detailed in this report. The technology used for development of VN Spacecraft is feasible and can be ready for January 2018 mars mission.**

Keywords: TPS,VN Spacecraft, Crew transfer Vehicle, Service module, Emergency Escape System, ECLSS

1. Introduction

A **Mars flyby** is a movement of spacecraft passing in the vicinity of the planet Mars, but not entering orbit or landing on it. Our spacecraft is designed for a flyby Mission for Planet Mars. The spacecraft they live in on the journey to Mars does the flyby, but the crew separates and goes into a lander. The Excursion module's ascent stage must rejoin the main spacecraft before it gets too far away. An advantage is that the resources needed for Earth return don't have to enter and leave Mars orbit, but the ascent stage has to perform space rendezvous in solar orbit and the time on Mars is constrained by the need to to this. Mars cyclers orbit the Sun in such a way as to pass by Mars and the Earth on regular intervals, performing Mars flybys on regular intervals. The crews would live on the stations during the interplanetary voyages. The concept for Flyby-Excursion Landing Module is that a lander and flyby would separate in solar orbit, the lander would accelerate to get to Mars first, then land on Mars meanwhile the other segment does a Mars flyby, then the lander takes off and rendezvous with the flyby segment transferring the crew over. Alternately, a flyby-only human mission is also possible, without detaching at Mars, but to slingshot around Mars and back to Earth.

As per Inspiration mars, They calculated an optimum trajectory launching in early January, 2018. At the Mars encounter, the spacecraft will pass within a few hundred kilometers of the surface. We compare a baseline SOA architecture with an advanced architecture. The advanced architecture uses recently developed equipment that has higher efficiencies for water recovery and lighter base mass. They are not currently in operation and therefore present a schedule risk for development and testing. the notional schedule based on state of the art

ECLSS technologies. ECLSS is a systems-integration-intense subsystem, so actual schedule is highly dependent on the vehicle integration schedule and timeline. The isolated, confined environment psychology aspects of the mission are considered with regard to crew selection, training, capsule design, the role of mission control / support, and early ground testing. We explore analogues such as Biosphere 2 and long duration spaceflight. shown that an ECLSS based on SOA technologies is feasible and can be ready for January 2018. A minimalist approach using existing technologies can be safely and robustly realized by utilizing spares and a crew capable of servicing and replacing the equipment.

The American manned Mars flyby. Study 1996. In 1996 Robert Zubrin proposed a new version of a manned Mars flyby mission, dubbed Athena.Unlike previous fly-by concepts, Athena would remain in the vicinity of Mars for a year while the crew remotely operated probes of the Martian surface and atmosphere. This would eliminate the round-trip radio time lag of ten to forty minutes in trying to operate such probes from the earth.

Athena would have a crew of two. The spacecraft and its trans-Mars injection stage would be assembled in low earth orbit using two shuttle launches and four Proton launches. The Mars probes (four rovers) would be launched separately by Delta 7925 or Molniya launch vehicles. Other possible probes would include subsonic remotely piloted drones or controlled balloons equipped with imaging systems and deployable/recoverable rovers.

Within the framework of the Mars DRA 5.0, a future block upgrade of the Orion CEV serves two vital functions: (1) the transfer of as many as six crew members between Earth and an MTV in LEO at the beginning of the Mars mission, and (2) the return of the as many as six crew members to Earth via

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direct entry from the Mars return trajectory. A CEV block upgrade (crew module and SM with a 3-year in-space certification) is launched as part of the crewed payload mass on an Ares V. The ISS version of the Orion, which will be launched by the Ares 1, delivers the six Mars crew members into an orbit that matches the inclination and altitude of the orbiting MTV. It then takes the CEV, which is conducting a standard ISS type rendezvous and docking approach to the MTV, as many as 2 days to perform orbit-raising maneuvers to close on the MTV. After docking, the CEV, the crew performs a leak check, equalizes pressure with the MTV, and opens hatches. Once crew and cargo transfer activities are complete, the crew delivery CEV is jettisoned in preparation for TMI. The long-lived Orion block upgrade that was delivered on the Ares V is configured to a quiescent state and remains docked to the MTV for the trip to Mars and back to Earth. Periodic systems health checks and monitoring are performed by the ground and flight crew throughout the mission. As the MTV approaches Earth upon completion of the 30-month round-trip mission, the crew performs a preundock health check of all entry-critical systems, transfers to the CEV, closes hatches, performs leak checks, and undocks from the MTV. The MTV is targeted for an Earth fly-by with subsequent disposal in heliocentric space. The CEV departs from the MTV 24 to 48 hours prior to Earth entry and conducts an on-board-targeted, ground-validated burn to target for the proper entry corridor; as entry approaches, the CEV CM maneuvers to the proper entry interface (EI) attitude for a direct-guided entry to the landing site. The CEV performs a nominal water landing, and the crew and vehicle are recovered. Earth entry speeds from a nominal Mars return trajectory may be as high as 12 km/s, as compared to 11 km/s for the lunar CEV. This difference will necessitate the development of a higher-density, lightweight TPS

2. Heavy lift launch vehicle

The launch vehicle selected for our Mars Mission is Angara-7V. Angara, named after a fast-flowing 1,800 km long Siberian river, should become Russia's first entirely post-Soviet space launch vehicle. Khrunichev State Research and Production Center is developing a family of Angara launchers to replace existing Proton and Rokot boosters.

The cash-starved, start-stop development effort officially began during the mid-1990s, but progress toward a planned inaugural launch in 2010-2011 has only recently become apparent.

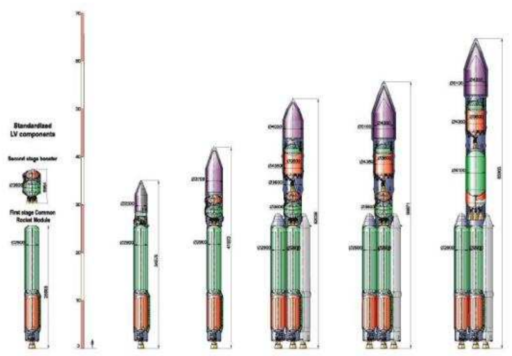


Figure 1 Angara launch vehicle variants

Angara will adapt several existing Russian space systems for its own use. A 30 tonne thrust LOX/kerosene RD-0124 engine will power the second stage of all but the smallest Angara version. This staged-combustion engine has already been developed to power the upgraded Soyuz-2 third stage. Briz-KM, developed for use on Rokot, will serve as the Angara 1.1 second stage and as the third stage for Angara 1.2. The Briz-M stage previously developed to fly atop Proton-M boosters will serve as the upper stage for the Angara 3 and 5 vehicles. The Rokot payload fairing will be used by Angara 1.1. Angara 3 and 5 will use Proton payload fairings. Normal Angara 3 and 5 launch profiles would involve keeping the core URM engine throttled back so that the strap on URMs would deplete their propellant first, about four minutes after liftoff. The core URM would then burn for another 89 to 111 seconds. Payload capabilities will extend from 2 metric tons (tonnes) to a 200 km x 63 deg low earth orbit (LEO) for Angara 1.1 to 24.5 tonnes for Angara 5 when launched from Plesetsk. Angara 5 will be able to boost 5.4 tonnes to geosynchronous transfer orbit (GTO) from Plesetsk with a Briz M upper stage. Use of the KVRB stage would improve GTO performance to 6.6 tonnes.

Characteristics	Angara-7	Angara-7/KRB	Angara-7P	Angara-7V	Angara A 7
Approximate date of the concept	2006	2006	2008	2008	2009
Payload mass in the low-Earth orbit*	35 tons	41 tons	36 tons	40.5 tons	35 tons***
Payload mass to the geostationary transfer orbit**	-	-	-	-	12.5 tons***
Payload mass in the geostationary orbit**	-	-	7.5 tons**		7.6 tons***
Relative mass of the payload	-	-	3.2 percent	3.51 percent	
Liftoff mass	1,122 tons	1,181 tons	1,125 tons	1,154 tons	1,133 tons
Payload fairing mass	-	-	3.5 tons	3.5 tons	
Payload fairing length	22 meters	26 meters			
Payload fairing diameter	6.5 meters	6.5 meters	5.5 meters	5.5 meters	
Emergency escape system mass	-	-	0.5 tons	-	
Number of stages	2	3	-	-	
Number of first stage boosters	6	6	-	-	

Stage I, II oxidizer	Liquid oxygen	Liquid oxygen	Liquid oxygen	Liquid oxygen	Liquid oxygen
Stage I, II fuel	Kerosene	Kerosene	Kerosene	Kerosene	Kerosene
Stage I propulsion	Six RD-191	Six RD-191	Six RD-191	Six RD-191	
Stage I total thrust on the Earth surface	1,176 tons	1,176 tons	1,176 tons	1,176 tons	
Stage I propellant mass	-	-	765.9 tons	765.9 tons	
Stage II propulsion	One RD-191	One RD-191	One RD-191	One RD-191	
Stage II total thrust in vacuum	216 tons	216 tons	216 tons	216 tons	
Stage II propellant mass	-	-	240 tons	240 tons	
Stage III oxidizer	-	-	-	Liquid oxygen	
Stage III fuel	-	-	-	Liquid hydrogen	
Stage III propulsion	-	-	-	Two RD-0146	
Stage III total thrust in vacuum	-	-	-	20 tons	
Stage III propellant mass	-	-	-	19.6 tons	

Table 1 Angara Launch vehicle Characteristics

*Circular orbit with the altitude of 200 km, and inclination 51.6 degrees toward the Equator**Requires the use of the KVRB upper stage

3. Performance characteristics of Angara-7V launch vehicle

Characteristics	**Angara-7V**
Approximate date of the concept	2008
Payload mass in the low-Earth orbit*	40.5 tons
Relative mass of the payload	3.51 percent
Liftoff mass	1,154 tons
Payload fairing mass	3.5 tons
Payload fairing diameter	5.5 meters
	-
Stage I, II oxidizer	Liquid oxygen
Stage I, II fuel	Kerosene
Stage I propulsion	Six RD-191
Stage I total thrust on the Earth surface	1,176 tons
Stage I propellant mass	765.9 tons
Stage II propulsion	One RD-191
Stage II total thrust in vacuum	216 tons
Stage II propellant mass	240 tons
Stage III oxidizer	Liquid oxygen
Stage III fuel	Liquid hydrogen
Stage III propulsion	Two RD-0146
Stage III total thrust in vacuum	20 tons
Stage III propellant mass	19.6 tons

4. Conceptual Design of VN Spacecraft

The **Crew and Service Module** (CSM) stack consists of two main parts: a conical Crew Module (CM), and a cylindrical Service Module (SM) holding the spacecraft's propulsion system and expendable supplies. It is designed to support long-duration deep space missions of up to four years. The spacecraft's life support, propulsion, thermal protection and avionics systems are designed to be upgradeable as new technologies become available. The VNCV spacecraft includes both crew and service modules, and a spacecraft adaptor.

The VNCV's crew module is larger than Apollo's and can support more crew members for short or long-duration spaceflight missions. The service module fuels and propels the spacecraft as well as storing oxygen and water for astronauts. The service module's structure is also being designed to provide locations to mount scientific experiments and cargo.

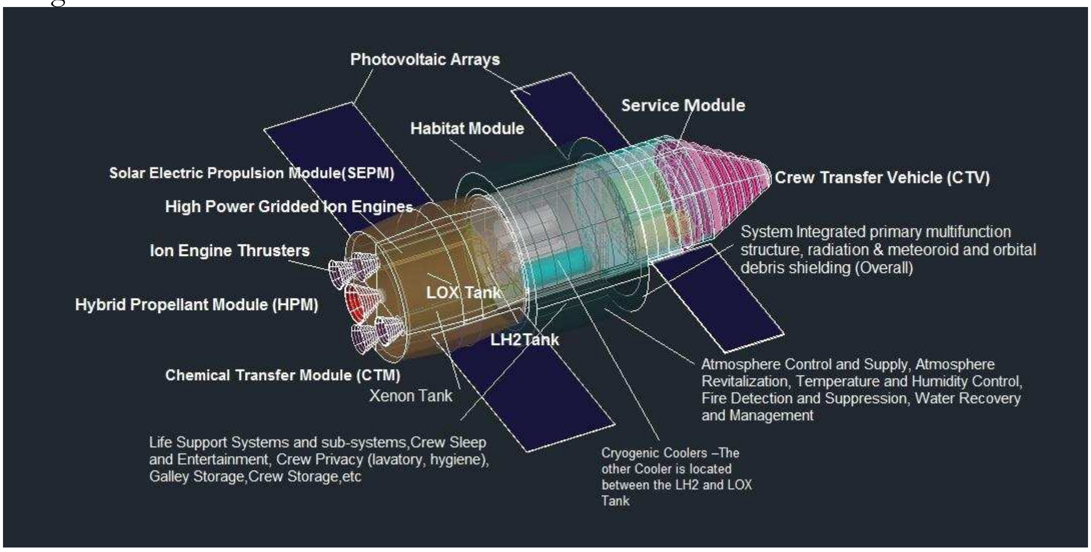


Figure.2 Detailed modules in Vimana Notion spacecraft

4.1 Crew Module :

The Vimana Notion CM will hold two crew members, compared to a maximum of three in the smaller Apollo CM or seven in the larger space shuttle. Despite its conceptual resemblance to the 1960s-era Apollo, Similar to Orion's CM will use several improved technologies, including:

- "Glass cockpit" digital control systems derived from that of the Boeing 787. An "autodock" feature, like those of Russian Progress spacecraft and the European Automated Transfer Vehicle, with provision for the flight crew to take over in an emergency. Previous American spacecraft (Gemini, Apollo, and Space Shuttle) have all required manual piloting for docking.
- Improved waste-management facilities, with a miniature camping-style toilet and the unisex "relief tube" used on the space shuttle (whose system was based on that used on Skylab) and the International Space Station (based on the Soyuz, Salyut, and Mir systems). This eliminates the use of the much-hated plastic "Apollo bags" used by the Apollo crews.
- A nitrogen/oxygen (N2/O2) mixed atmosphere at either sea level (101.3 kPa or 14.69 psi) or slightly reduced (55.2 to 70.3 kPa or 8.01 to 10.20 psi) pressure.
- Much more advanced computers than on previous manned spacecraft.

Another feature will be the partial reusability of the Vimana Notion CM. Both the CM and SM will be constructed of the aluminium lithium (Al/Li) alloy like that was used on the shuttle's external tank.. The CM itself will be covered in the same Nomex felt-like thermal protection blankets used on parts on the shuttle not subject to critical heating, such as the payload bay doors. The reusable recovery parachutes will be based on the parachutes used on both the Apollo spacecraft and the Space Shuttle Solid Rocket Boosters, and will also use the same Nomex cloth for construction. Water landings will be the exclusive means of recovery for the Vimana Notion CM.

Both the spacecraft and docking adapter will employ a Launch Escape System (LES) like that used in Mercury and Apollo, along with an Apollo-derived "Boost Protective Cover" (made of fiber glass), to protect the Vimana Notion CM from aerodynamic and impact stresses during the first 2 ½ minutes of ascent.

The Vimana Notion Crew Module (CM) is a 52° frustum shape, similar to that of the Apollo Command Module. As projected, the CM will be 5.02 meters (16 ft 6 in) in diameter and 3.3 meters (10 ft 10 in) in length, with a mass of about 8 metric tons (19,000 lb). It will have more than 50% more volume than the Apollo capsule, which had an interior volume of 5.9 m^3 (210 cu ft), and will carry two to six astronauts. After extensive study, NASA has selected the Avcoat ablator system for the Orion crew module. Avcoat, which is composed of silica fibers with a resin in a honeycomb made of fiberglass and phenolic resin, was previously used on the Apollo missions and on select areas of the space shuttle for early flights.

The crew module is the transportation capsule that provides a habitat for the crew, provides storage for consumables and research instruments, and serves as the docking port for crew transfers. The crew module is the only part of the MPCV that returns to Earth after each mission.

The crew module will have 316 cubic feet (8.9 m^3) and capabilities of carrying four astronauts for 21 day flights itself which could be expanded through additional service modules. Its designers claim that the MPCV is designed to be 10 times safer during ascent and reentry than the Space Shuttle.

4.2 Service Module

Vimana Notion Service Module serves as the primary power and propulsion component of the Orion spacecraft, but can be discarded at the end of each mission. It provides in-space propulsion capability for orbital transfer, attitude control, and high altitude ascent aborts. When mated with the crew module, it provides the water and oxygen needed for a habitable environment, generates and stores electrical power while on-orbit, and maintains the temperature of the vehicle's systems and components. This module can also transport unpressurized cargo and scientific payloads.

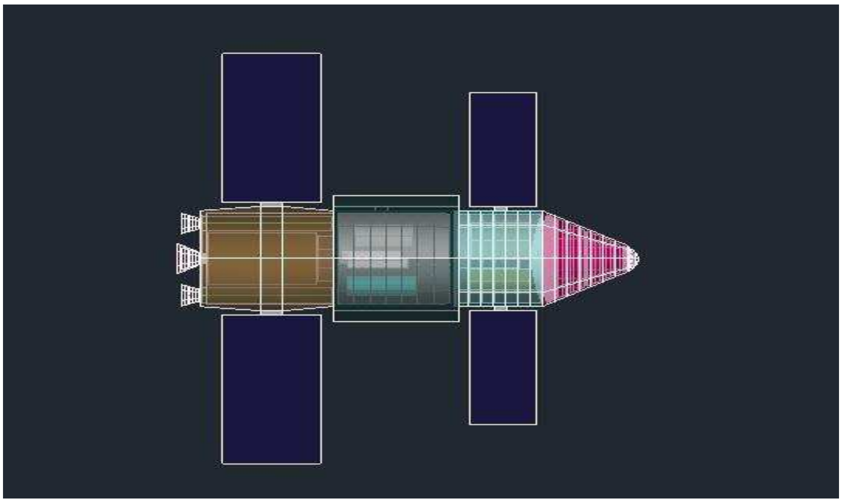


Figure.3 Detailed systems in Vimana Notion spacecraft

Roughly cylindrical in shape, the Orion service module, like the crew module, will be constructed of Al-Li alloy (to keep weight down), and will feature a pair of deployable circular solar panels, similar in design to the panels used on the Mars Phoenix lander.

The panels, the first to be used on a U.S. manned spacecraft (except for a 10-year period, the Soviet/Russian Soyuz spacecraft has used them since the first mission in 1967), will allow NASA to eliminate the need to carry malfunction-prone fuel cells, and its associated hardware (mainly LH_2 tanks) from the service module, resulting in a shorter, yet more maneuverable spacecraft. The Successful Orion solar array design using full-scale "UltraFlex wing" hardware is used in Vimana Notion Spacecraft.

The Main Engine (ME) is a 8500 pound thrust, pressure-fed, regeneratively cooled, storable bi-propellant rocket engine made by Aerojet. The ME is an increased performance version of the 6000-pound thrust rocket engine used by the Space Shuttle for its Orbital Maneuvering System (OMS). The SM Reaction Control System (RCS), the spacecraft's maneuvering thrusters (originally based on the Apollo "quad" system, but currently resembles that used on Gemini), will also be pressure-fed, and will use the same propellants. NASA believes the SM RCS would be able to act as a backup for a trans-Earth injection (TEI) burn in case the main SM engine fails.

4.3 Pre-ATV Service Module design

A pair of LOX tanks (similar to those used in the Apollo SM) will provide, along with small tanks of nitrogen, the crew with breathing air at sea-level or "cruising altitude" pressure (14.7 or 10.2 psi), with a small "surge tank" providing necessary life support during reentry and touchdown. Lithium hydroxide (LiOH) cartridges will recycle the spacecraft's environmental system by "scrubbing" the carbon dioxide (CO_2) exhaled by the astronauts from ship's air and adding fresh oxygen and nitrogen, which is then cycled back out into the system loop. Because of the switch from fuel cells to solar panels, the service module will have an onboard water tank which will provide drinking water for the crew, and (when mixed with glycol), cooling water for the spacecraft's electronics. Unlike the practice during Apollo of dumping both water and urine overboard during the flight, the Orion will have an onboard recycling system, identical to that used on the International Space Station, that will convert both waste water and urine into both drinking and cooling water.

The Service Module also mounts the spacecraft's waste heat management system (its radiators) and the aforementioned solar panels. These panels, along with backup batteries located in the Vimana Notion CM, will provide in-flight power to the ship's systems. The voltage, 28 volts DC, is similar to that used on the Apollo spacecraft during flight.

The Vimana Notion service module would be encapsulated by fiberglass shrouds jettisoned at the same time as the LES/Boost Protective Cover, which would take place roughly 2½ minutes after launch (30 seconds after the solid rocket first stage is jettisoned). Vimana Notion service module design that allows in to make the vehicle lighter in weight and permitting the attachment of the circular solar panels at the module's midpoints, instead of at the base near the spacecraft/rocket adapter, which may subject the panels to damage.

The Vimana service module (SM) is projected comprising a cylindrical shape, having a diameter of 5.03 m (16 ft 6 in) and an overall length (including thruster) of 4.78 m (15 ft 8 in). With solar panels extended, span is either 17.00 m (55.77 ft) or 55.00 ft (16.76 m) The projected empty mass is 3,700 kg (8,000 lb), fuel capacity is 8,300 kg (18,000 lb).

4.4 Emergency escape module in Space

In the event of an emergency on the space during space travel, the service module with auxiliary will separate the Crew Module from the space vehicle using a powered launch abort motor (AM), which is more powerful than the Atlas 109-D booster that launched astronaut John Glenn into orbit in 1962. There are two other propulsion systems in the LAS stack: the attitude control motor (ACM) and the jettison motor (JM). On July 10, 2007, Orbital Sciences, the prime contractor for the LAS, awarded Alliant Techsystems (ATK) a $62.5 million sub-contract to, "design, develop, produce, test and deliver the launch abort motor." ATK, which had the prime contract for the first stage of the Ares I rocket, intended to use an innovative "reverse flow" design for the motor. On July 9, 2008, NASA announced that ATK had completed a vertical test stand at a facility in Promontory, Utah to test launch abort motors for the Orion spacecraft. Another long-time space motor contractor, Aerojet, was awarded the jettison motor design and development contract for the LAS. As of September 2008, Aerojet has, along with team members Orbital Sciences, Lockheed Martin and NASA, successfully demonstrated two full-scale test firings of the jettison motor. This motor is important to every flight in that it functions to pull the LAS tower away from the vehicle after a successful launch. The motor also functions in the same manner for an abort scenario.

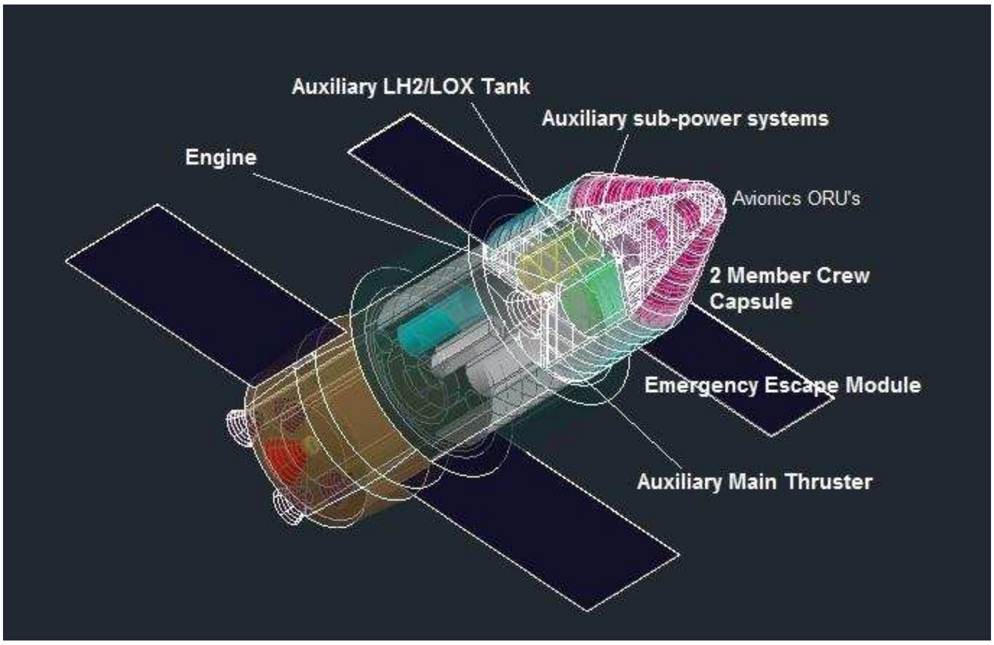


Figure 4 Emergency escape module in Space

4.5 Hybrid Propellant Module

A Hybrid Propellant Module (HPM) that combines both chemical and electrical propellant in conjunction with modular orbital transfer/engine stages was targeted as the core element. The HPM incorporates zero boil-off technology to maintain its cryogenic propellant load for long periods of time. The fundamental concept for an HPM-based in-space transportation architecture requires two HPMs and two propulsive transfer stages: one chemical-based and one electric-based. The basic philosophy is to utilize the chemical propellant stored onboard the HPM in conjunction with a chemical transfer/engine stage to provide high thrust during the time critical segments of a mission (e.g., crew transfers). The electric propellant is utilized with a solar electric transfer/engine stage during non-time critical segments of the mission (e.g., prepositioning an HPM for the crew return segment of the mission, and return of an HPM to its parking orbit). This architecture can save a significant amount of propellant when compared to an all chemical mission assuming that the efficiency of the electric propulsion system is sufficiently greater than the chemical ropulsion system. Chemical engines that use liquid oxygen (LOX) and liquid hydrogen (LH_2) are assumed to have a specific impulse (Isp) of 466 seconds. Electrical propulsion engines using xenon propellant are assumed to have an Isp of 3,000 seconds or greater. Although chemical propellant is still required for each crew transfer segment of the mission, the mass penalty for carrying the return trip chemical propellant is substantially reduced due to the substantially higher specific impulse of the electric propulsion system.

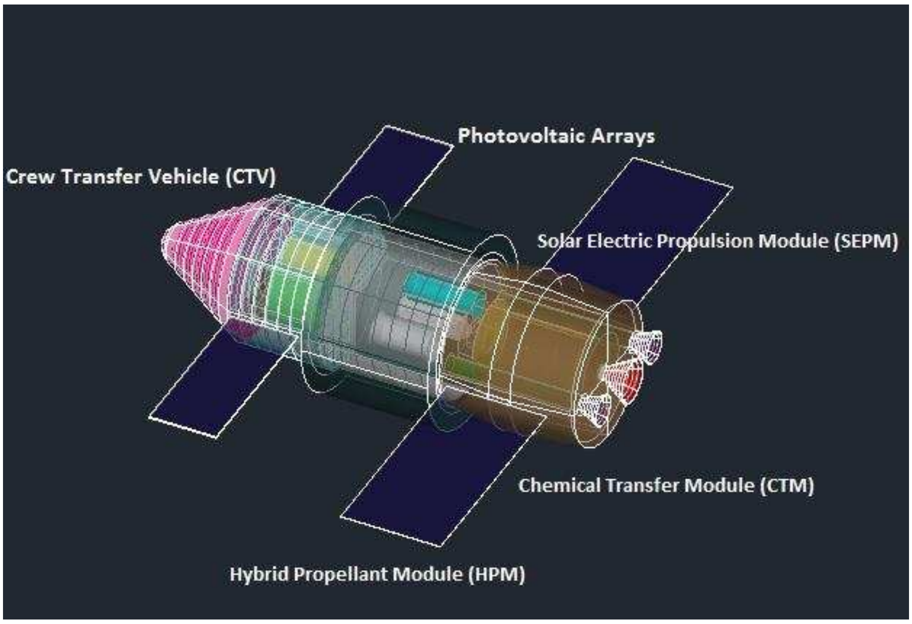


Figure 5 crew transfer module

The principal driver for the HPM configuration is the requirement for launch by a Shuttle-class vehicle. For Shuttle compatibility the HPM is restricted to a length of 14.2 m, a diameter of 4.5 m, and a maximum mass of 14.5 MT. The HPM configuration is divided into an upper section with a maximum diameter of 4.5 m and a lower section with a maximum diameter of 4.0 m. The smaller diameter of the lower section allows the PV arrays, body mounted radiators and ORUs to be stowed along the HPM within the diameter constraints of the Shuttle payload bay. The HPM upper and lower sections are tapered to better transfer loads. Since the HPM will at times be flown and maintained in LEO, micrometeoroid and orbital debris (MMOD) shielding is required. The HPM upper section design incorporates an expandable (10 cm compacted, 30 cm expanded) multi-shock shield that is expanded at HPM deployment. Use of an expandable MMOD design for the HPM upper section allows for maximum diameter of the HPM primary structure within the Shuttle payload bay constraints. Due to packaging constraints and complications involved with deploying an expandable MMOD shield around the PV array arms, radiators and orbital replaceable units (ORUs), a non-expandable syntactic aluminum foam is used for MMOD shielding on the HPM lower section. A combined standoff distance of 30 cm was determined to be adequate between the primary structure and MMOD shielding The maximum requirements for LH_2 and LOX were determined to be 4,450 kg and 26,750 kg, respectively. This gives a total chemical propellant mass of 31,200 kg. The internal volume required for the LH_2 and LOX tanks was thus found to be 66 m^3 and 24 m^3, respectively. The maximum requirement for LXe was found to be 13,600 kg, requiring an internal tank volume of 4 m^3. Since the density of LOX and LXe is considerably greater than that of LH_2, these tanks are located as close to the propulsion module interface as possible in order to maintain the HPM center of gravity (CG) as far aft as possible. The HPM aft CG is necessary for controllability during HPM operations and to potentially meet CG constraints of the Shuttle-class launch vehicle. The larger upper section of the HPM is used to accommodate the larger volume of LH_2. The LOX tank is placed directly adjacent to the LH_2 tank to utilize the same cryogenic cooling system. A single LXe tank utilizes a tapered, conical shape to maximize available tank volume.

4.6 Chemical Transfer Module

The Chemical Transfer Module (CTM) serves as a high energy injection stage when attached to an HPM and an autonomous orbital maneuvering vehicle for proximity operations such as ferrying payloads a short distance, refueling and servicing. It has high thrust H_2O_2 engines for orbit transfers an high-pressure H_2O_2 thrusters for proximity operations and small delta-V translational or rotational maneuvers. It is capable of transferring and storing approximately 3,000 kg of cryogenic hydrogen and oxygen. The main engines can use the stored cryogens or utilize propellant directly transferred from the HPM. Unlike the HPM, the CTM does not incorporate zero boil-off technology. The CTM deployed length is approximately9.4 meters. The CTM width, with solar arrays deployed, is approximately 12.6 meters.

The major components of the CTM are:
· Dual RL10 67 kN-class engines
· Liquid oxygen (LOX) tank
· Liquid hydrogen (LH_2) tank
· Gaseous oxygen (GOX) RCS tank
· Six gaseous hydrogen (GH_2) RCS tanks
· Two deployable solar arrays
· Avionics modules
· Two radiator panels
· Four sets of tri-pod RCS thrusters
· Four sets of tri-pod cold gas thrusters
· Docking adapter

The dual RL10 engines are mounted twenty degrees off the CTM centerline on a fixed thrust structure. Two engines are required to satisfy reliability requirements. Since only one engine is used at a time, the thrust structure and the two engines are rotated as a single unit such that the firing engine thrust vector is aligned with the vehicle center of gravity. A new development gimbal system is required to accomplish this operation Two sets of tri-pod RCS thrusters and two sets of tri-pod cold gas thrusters are mounted on the aft end of the CTM. The thruster pods are all canted forty-five degrees to avoid plume impingement on the CTM thrust structure MMOD shield. Two sets of tri-pod RCS thrusters and two sets of tri-pod cold gas thrusters are mounted on the forward end of the CTM. These thrusters pods are mounted on fixed booms and canted forty-five degrees to prevent plume impingement on an attached HPM. MMOD shielding encloses the CTM tankage and plumbing to satisfy safety requirements. The avionics ORUs are packaged in the forward skirt to avoid the adverse thermal environment in the vicinity of the RL10 engines.

4.7 Solar Electric Propulsion Stage

The Solar Electric Propulsion (SEP) Stage serves as a low-thrust stage when attached to an HPM for pre-positioning large and/or massive elements or for the slow return of elements to LEO for refurbishing and refueling.

The SEP Stage is comprised of three elements:
· Thruster Pallet
· Deployable Boom
· Base Pallet

The Thruster Pallet is a circular plate used to mount multiple electric thrusters on lightweight gimbals. The gimbals are incorporated to enable small pointing corrections to offset any beam aberrations in each thruster. A power processing unit (PPU), one per thruster, converts input power from the arrays into the required thruster power. A gas distribution unit (GDU), located on the thruster face of the pallet, serves as a manifold for propellant delivery to the thrusters. Each engine includes a propellant feed system that regulates input flow as required for engine operation. A loop heat pipe system is mounted on the Thruster Pallet to reject waste heat from the Power Processing Units (PPUs). The rejected heat is conducted to two radiator wings attached to the Thruster Pallet.

The Thruster Pallet is attached to the Deployable Boom. This boom enables the Thruster Pallet to be articulated over large angles while the Base Pallet and the HPM are maintained in a solar inertial attitude for solar array pointing. The Thruster Pallet position is continually adjusted to maintain a relatively constant thrust vector through the spacecraft center of mass in order to maximize effective thrust. The Deployable Boom also provides sufficient distance between the thrusters and the solar arrays to prevent degradation due to exhaust plume impingement and erosion.

The Base Pallet houses the solar arrays and associated power management and distribution components, the docking mechanism and fluid transfer interfaces, and other systems. This pallet is a cylindrical structure with a rigid boom attached at the center of one face. On the opposite face is the docking mechanism and fluid transfer interface which mate with the HPM. Two large, rectangular-shaped solar arrays are attached to the Base Pallet sides. These arrays are on stand-off booms to provide the necessary clearance with the HPM structure.

The solar arrays consist of advanced, thinfilm cells on a lightweight substrate supported on a collapsible, cell-structure wing architecture. This architecture has the advantage of packing very compactly and does not impose size limitations impacting launch vehicle manifesting. The arrays are required to accommodate a one-time deployment only.Other elements inside the Base Pallet include:

· A gas distribution unit to handle xenon flow through the pallet from the HPM to the thrusters

· A reaction wheel-based system for attitude control during electric thruster operation

· A Guidance Navigation & Control(GN&C) unit

· A Command & Data Handling (C&DH) unit

· A battery system to power deployment of the solar arrays

· A xenon tank loaded with 2,000 kg of xenon for free- flying operations during SEP Stage orbital parking

· A Reaction Control System (RCS) for docking maneuvers consisting of four thruster pods and two propellant tanks (containing gaseous hydrogen and gaseous oxygen)

· A Thermal Control System (TCS) comprised of two radiator wings attached to the outside of the base pallet and a loop heat pipe system mounted inside that conducts waste heat from the Power Processing Units (PPUs).

5. Communication systems:

We propose the NASA Deep Space Network (DSN),an international network of antennas to provide the communication links between the scientists and engineers on Earth to the Mars Exploration Rovers in space and on Mars. The DSN consists of three deep-space communications facilities placed approximately 120 degrees apart around the world. This strategic placement permits constant observation of spacecraft as the Earth rotates on its own axis.

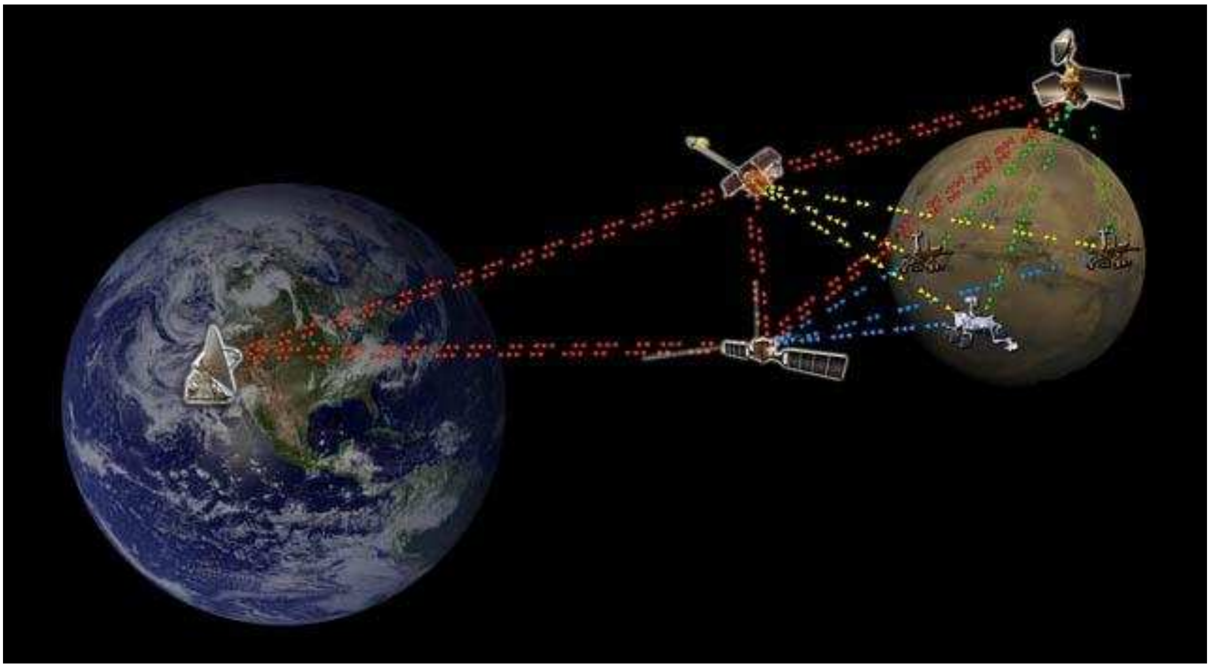


Figure 6 first deep space communications network modelled by NASA

5.1 How the spacecraft can communicate through Mars-orbiting spacecraft

Not only can the rovers send messages directly to the DSN stations, but they can uplink information to other spacecraft orbiting Mars, utilizing the 2001 Mars Odyssey and Mars Global Surveyor orbiters as messengers who can pass along news to Earth for the rovers. The orbiters can also send messages to the rovers.

The benefits of using the orbiting spacecraft are that the orbiters are closer to the rovers than the DSN antennas on Earth and the orbiters have Earth in their field of view for much longer time periods than the rovers on the ground.Because the orbiters are only 250 miles (400 kilometers) above the surface of Mars, the rovers don´t have to "yell" as loudly (or use as much energy to send a message) to the orbiters as they do to the antennas on Earth. The distance from Mars to Earth (and from the rovers to the DSN antennas) during the primary surface missions varies from 110 to 200 million miles (170 to 320 million kilometers).

5.2 Data speed and volume:

The data rate direct-to-Earth varies from about 12,000 bits per second to 3,500 bits per second (roughly a third as fast as a standard home modem). The data rate to the orbiters is a constant 128,000 bits per second (4 times faster than a home modem). An orbiter passes over the rover and is in the vicinity of the sky to communicate with the rovers for about eight minutes at a time, per sol. In that time, about 60 megabits of data (about 1/100 of a CD) can be transmitted to an orbiter. That same 60 megabits would take between 1.5 and 5 hours to transmit direct to Earth. The rovers can only transmit direct-to-Earth for at most three hours a day due to power and thermal limitations, even though Earth may be in view much longer.Mars is rotating on its own axis so Mars often "turns its back" to Earth, taking the rover with it. The rover is turned out of the field of view of Earth and goes "dark", just like nighttime on Earth, when the sun goes out of the field of view of Earth at a certain location when the Earth turns its "back" to the sun. The orbiters can see Earth for about 2/3 of each orbit, or about 16 hours a day. They can send much more data direct-to-Earth than the rovers, not only because they can see Earth longer, but because they can operate their radio for much longer since their solar panels get light most of the time, and they have bigger antennas than the rovers.

6. Trajectory and Launch

1. The launch dates as per the Inspiration Mars mission is stated to be January 5th, so I have taken that as the starting point. The trajectory would be as simple as possible, using chemical to break Earth orbit and exit the Earth SOI, then enter into a hyperbolic orbit with Mars at a focus and have a perigee of 100km, as stated in the problem statement. The hyperbolic nature of the orbit, would give the ship a gravity boost, that would increase the speed, but Earth by that time would not be in a proper reception position, so a Venus flyby is inevitable. This calls for a mission having nearly 210-260

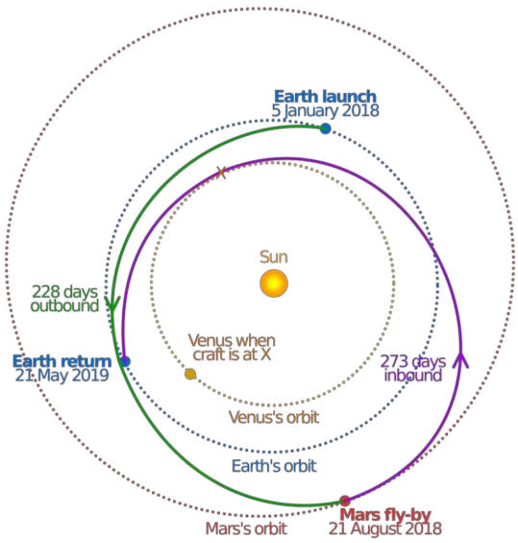


Figure 7 Inspiration Mars Trajectory

2. The trajectory of the mission calls for an endurance of over a year, which means that the consumables meant to supply the 2 astronauts for the entire duration of the mission has to last over 230 + 280 days. Secondly, two people being confined to a small volume of space, will eventually go 'stir crazy', especially coupled with the fact that there can be no rescue in the duration. Thus the design of the crew vehicle

days for a Earth-Mars transit. and an ~ 280 day return journey.

Sticking to this flight profile (as provided in the Inspiration Mars website) seems the best option. As we are not allowed to use any nuclear powered propulsion unit, the only options we are left with are either solar powered Ion thrusters or chemical.
Disadvantage of ion unit - The current ion units produce very high I_{sp} but require enormous amounts of energy to do so. Secondly they are in the low thrust category. So using an ion engine and its associated fuel, provides for a weight and power penalty and does not reap any discernible benefits to the trajectory (in terms of reduction of transit time). A full Ion based drive system for Earth to Mars transit would require almost 1000 days!!. So my suggestion is to forgo the ion system and use standard chemical systems. The fuel can either be CH_4 - LOX or LH_2 – LOX
becomes an important factor, and in this case a spacious vehicle is needed. There was a study that the Soviet Union conducted back in the 1960s known as the TMK or *Tyazhely Mezhplanetny Korabl* for *Heavy Interplanetary Spacecraft*. Check out this link - http://en.wikipedia.org/wiki/TMK. It called for several heavy lift rockets to assemble an interplanetary spacecraft in Mars orbit, housing a habitation/pilot, work/equipment,biological,aggregate/engineering and an Earth return capsule. This is very much the concept that is needed now. Unfortunately the system was to be assembled in Low Earth Orbit using the doomed N1 launch vehicles requiring upto 25 launches to fully assemble. But the assembled spacecrafts would have a variety of endurance capabilities depending upon the volume of available to the crew members. For the smallest variation, the Russians had chosen a crew of three. In the modern context, it would be very feasible to adapt this concept as a low cost yet highly feaseable approach.

Some of the advantages in the modern context of the current mission would be the following:
(a) Light weight and modular construction materials to reduce the weight of the modules being lifted into Earth orbit.
(b) Some of the components that may be used for this mission, can be brawn from existing and highly proven technologies of Mir and ISS.
(c) It is also given in the problem definition that the Inspiration Mars intends to send the Inflatable habitat module. So modularity is a key to the design of the crew vehicle, as the technology for the habitat is still undergoing tests and may or may not be ready by the prescribed launch date.
Modularity was not a key factor in the design of the TMK missions. In light of the above my suggestion is to use modules similar to the ones used on Mir / ISS. Please have a look at the following description to gain an overview of what the crew vehicle might look like:
There will also be several sections to the crew vehicle:
Pilot Module: The astronauts will control the spacecraft from this module, while also conducting several experiments with respect to astronomical phenomenon. This should possibly be a Modified 'Zarya' Module. It will contain two solar panels like the original. There will also be a docking ball, for addition of other modules.

Habitation module: This module will also be the size of the Zarya module, where the astronauts will live and conduct day to day recreational activities. The orientation of this module will be parallel to the original pilot module.

Earth Return Module: This can either be a Soyuz or an Orion spacecraft that will be used for the crew to return to the surface of the Earth.

Payload Module: The payload module will contain the palettes, that will be dropped onto the surface of Mars to conduct experiments on the surface. The section containing the palettes will be unpressurized. But half of the module will be pressurized and will contain the consumables required for the astronauts. This module will be attached to the docking ball, at a right angle to the axis of travel.

This leaves almost three other openings on the docking ball, that can be used as necessary to carry more equipment to be dropped onto the surface of Mars. The presence of the docking port also allows this vehicle to accommodate the inflatable habitat that is meant to travel to the surface of Mars.
Engineering Module: The Engineering module contains a cluster of 3 LOX-CH_4/LOX-RP-1 engines, the fuel tanks, the solar panels and a high gain antennae. The positioning of this module can be determined according to the structure of the craft so as to which modules will actually be travelling.

Such modularity allows the mission developers to maximize the tech demonstration capability as well as scientific returns, while optimizing the cost. The minimum is three modules mentioned above (not taking into account the payload module), and hence three surface launches. The maximum is seven launches including the section for Engineering. Moreover. The ship once returned from one trip, can be parked in low earth orbit, and serve as the core module for the next mission, drastically reducing on cost.

3. To lift the modules into low earth orbit, it is also necessary to choose the proper launch mechanism. It is quite evident that the modules will be heavy, so heavy to ultra-heavy lift launch vehicles are required. Some of the existing, decommissioned and those in development in the U.S., Russia and India are as given:

Soyuz Fregat/2 (Medium) - 7000 - 7800 kg. - Operational - Russia
Proton (Heavy) - 22000kg. - Operational - Russia

Zenit (Medium) - 13000 kg. - Operational - Russia
Energia (Ultra-Heavy) - 100000 kg. - Decommissioned 1980 - Russia
Angara A3/A5/A7 (Medium - Heavy) - 10000-40000 kg. - Under Development - Russia
PSLV (Medium) - 3000-4000 kg. - Operational - India
GSLV Mk. II (Medium) - 5000 kg. - Operational/Proof Testing - India
GSLV Mk. III (Medium) - 10000 kg. - Under Development - India
Atlas III / V (Medium) - 6000 - 18000 kg. - Operational - U.S.
Delta IV/IV H (Medium - Heavy) - 10000 - 23000 kg. - Operational / Retired - U.S.
Falcon 9 (Medium) - 13000 kg. - Operational - U.S.
Falcon heavy (Heavy - Ultra-Heavy) - 53000 kg. - Under Development - U.S.
Saturn 1B/V (Medium - Ultra Heavy) - 12000 - 110000 kg. - Retired - U.S.
Space Shuttle - (Heavy) - 25000 kg. - Decommissioned 2011 - U.S.
Titan IV (Heavy) - 22000 kg. - Retired - U.S.
Space Launch System (Heavy - Ultra Heavy) - 70000 - 130000 kg. - Under Development - U.S.

Since it is required that the mission be launched, the given problem will have a higher confidence of solution if existing operational launchers with high success rates are used. As stated earlier a minimum of three launches are required to assemble the ship. The following scenarios are possible:

Scenario 1:
Proton Cargo Launch 1 (Proton K): Carries the Pilot module
Proton Cargo Launch 2 (Proton K): Carries the Habitation Module
Proton Cargo Launch 3 (Proton K): Modified upper stage carries the Engineering module with the Fuel necessary.bSoyuz Manned Launch (Soyuz-U): Carries the Soyuz vehicle and the crew.

Scenario 2:
Zenit Cargo Launch 1 (Zenit-2): Carries the Pilot module
Zenit Cargo Launch 2 (Zenit-2): Carries the Habitation Module
Proton Cargo Launch 3 (Proton K/Block DM): Modified upper stage carries the Engineering module with the Fuel necessary.
Soyuz Manned Launch (Soyuz-U): Carries the Soyuz vehicle and the crew.

Scenario 3:
Proton Cargo Launch 1 (Proton K): Carries the Pilot module
Proton Cargo Launch 2 (Proton K): Carries the Habitation Module
Delta Cargo Launch (Delta IV-H): Modified upper stage carries the Engineering module with the Fuel necessary.
Soyuz Manned Launch (Soyuz-U): Carries the Soyuz vehicle and the crew.

Scenario 4:
Falcon 9 Cargo Launch 1 (Falcon 9 v1.1): Carries the Pilot module
Falcon 9 Cargo Launch 2 (Falcon 9 v1.1): Carries the Habitat module
Falcon 9 Cargo Launch 3 (Falcon 9 v1.1 - modified): Modified upper stage carries the Engineering module with the Fuel necessary.
Falcon 9 Manned Launch (Falcon 9 v1.1): Carries the manned Dragon spacecraft and the crew.

Scenario 5:
Shuttle Derived Cargo & Manned 1: Launch the modified shuttle with crew habitation and payload into LEO
Proton Cargo Launch 3 (Proton K): Modified upper stage carries the Engineering module with the Fuel necessary.

Comparison of Scenarios (without any scientific payload capacity)

Scenario	Scenario 1	Scenario 2	Scenario 3	Scenario 4	Scenario 5
Estimated Total cost of launches	350 million $	250 million $	420 million $	226 million $	400 million $
Advantages	Proven parts and launch vehicles	Proven parts and launch vehicles. Lower cost	Proven parts and launch vehicles, with excess fuel	Uses the Falcon launch systems favoured by Inspiration Mars. Also minimum cost	Least complex, lowest no.of launches.and system is composite as everything is within the Shuttle itself. Also shuttle is also capable of modification. The launch infrastructure for the STS already exists and reactivation would require the least resources. This also has space for the scientific payload capacity, within that cost.
Disadvantages	Slightly costly and fairly complex in assembly. No scientific payload	The Zenit Launch system may force the use of less volume for the habitation modules, to keep the weight down. Also fairly complex to assemble. No scientific payload	Highest cost factor, also fairly complex to assemble. No scientific payload	The lower cost may force the use of smaller volume for the modules and less fuel. No scientific payload	Cost factor.

Table 2 Comparison of Scenarios

So reactivation also includes the redevelopment of the infrastructure required to support them. On the other hand, the Shuttle was retired in 2011, and the infrastructure is necessary for the next generation of Heavy lift launch vehicles, hence reactivation of the STS would require the least resources.

7. Power Subsystem

The power system on board the spacecraft will be based on three different sources. Since nuclear electric power sources are both expensive and may pose a health hazard to the crew, solar photovoltaic and solar thermal dynamic are considered as the only alternative. The use of nuclear sources, although compact and of minimal weight may suit the purpose of this mission, it also poses several health hazards and as of yet cannot generate power in the required amount to power all of the systems. Thus the use of both Solar photovoltaic and solar thermal dynamic remains as options.

Design Parameter	**Solar Photovoltaic**	**Solar Thermal Dynamic**	**Radio-isotope Thermoelectric**	**Nuclear Reactor**
Power Range (kW)	0.2-25	1-300	0.2-10	25-100
Specific Power (W/kg)	26-100	9-15	8-10	15-22
Specific cost ($/W)	2500-3000	800-1200	16k-18k	400-700
Hardness				
- Natural Radiation	Medium	High	Very High	Very High
- Nuclear threat	Medium	High	Very High	Very High
- Laser threat	Medium	High	Very High	Very High
- Pellets	Low	Medium	Very High	Very High
Stability and Maneuverability	Low	Medium	High	High
Degradation over Life	Medium	Medium	Low	Low
Storage required for Solar eclipse	Yes	Yes	No	No
Sensitivity to sun angle	Medium	High	None	None
Sensitivity to Spacecraft	Low	High	None	None

shadowing				
Obstruction of Spacecraft viewing	High	High	Low	Medium
Fuel Availability	Unlimited	Unlimited	Very low	Very low
Safety Analysis Reporting	Minimal	Minimal	Routine	Extensive
IR Signature	Low	Medium	Medium	High
Principal Applications	Earth Orbiting Spacecraft	Interplanetary	Interplanetary, Earth orbiting spacecraft	Interplanetary

Table 3 The following table summarizes the different forms of power sources available for the missions

Solar photovoltaic power sources or Solar panels are a very mature technology when it comes to space application. Almost all satellites use solar panels for power generation. But all previous manned missions beyond LEO has used stored power in the form of Fuel cells (namely the Apollo missions). All missions in LEO uses solar panels as a viable source of power (ISS, Soyuz).For the current mission, it is required to generate a large amount of energy, for driving the following subsystems simultaneously:

(1) Life Support.

(2) Radiation Shielding. (concept explained later.)

(3) Guidance, Navigation and Control systems.

(4) Propulsion.

(a) The use of an advanced Xenon engine dictates the requirement of copious amounts of electrical power to ionize and accelerate the ions.

(b) Use of chemical engines requires burst power.

(5) Energy Storage.

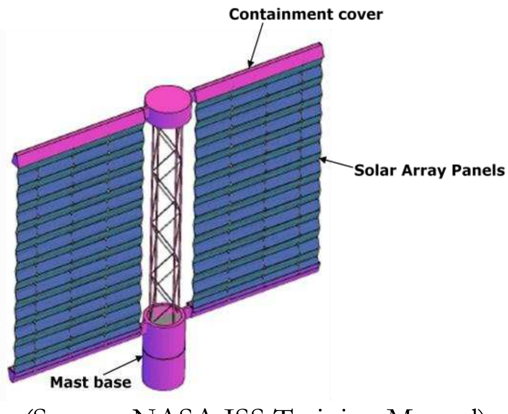
7.1 Solar Photovoltaic panels

The panels would have to be of sufficiently large area to generate approximately 50kW. But for launch they would have to be folded. The folding and extension mechanism can be the same as that for the ISS. The following picture shows one of the solar array panels partially deployed or under deploying conditions.

Assuming that the ship is powered by a 250kW DS4G Xenon Ion Engine that is currently under development, would mean the use of solar panels having an area that rivals that present on the ISS (The 8 solar panels on the ISS develops almost 220kW), and the requirement would be for the propulsion system alone.

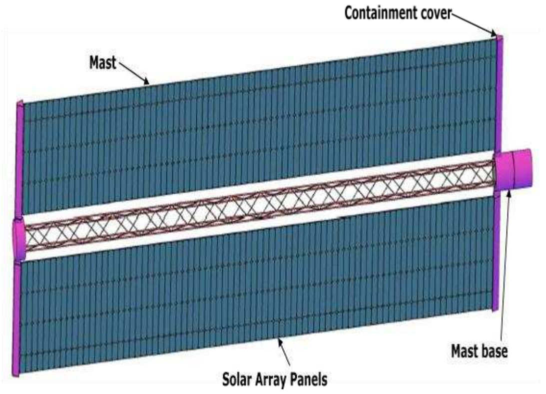
Thus to meet the power requirement of the mission, two separate yet tested methods are hereby proposed:

- Solar Panels - Standard multi-junction solar cells with Fresnels' concentrators for enhanced efficiency (Power output is approximately 50-75kW)
- Two Solar thermal units mounted on outboard arms for powering the radiation shields.
- Radiation Shielding panel units - converts cosmic radiation, high energy solar particles, UV, X-Ray and gamma rays into useful energy, and also mitigating the radiation threat.

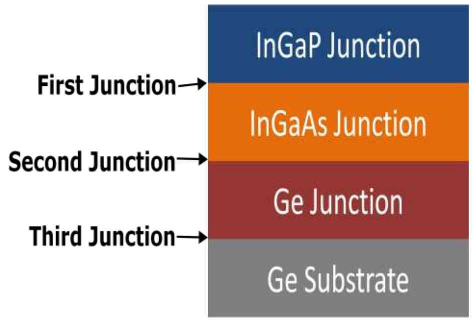


(Source: NASA ISS Training Manual)
The solar array panel partially deployed

The following picture shows the fully extended solar panel array.

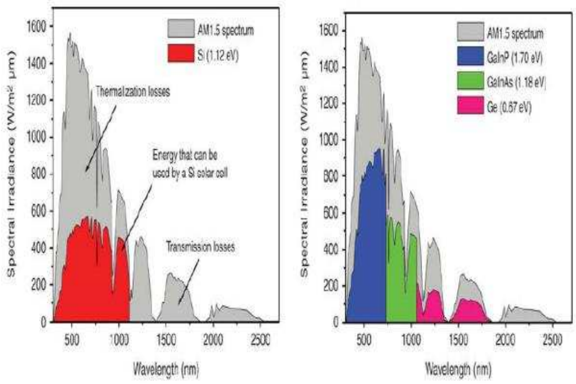


The photovoltaic solar cells are triple junction solar cells like (ZTJ Photovoltaic solar cell from emcore) The ZTJ cell structure is as shown in the following figure:



(Source: emcore ZTJ photovoltaic cell structure datasheet)

The following figure shows the solar and the terrestrial spectrum, and the advantage of using multijunction cells over single junction cells:



(Source: [1])

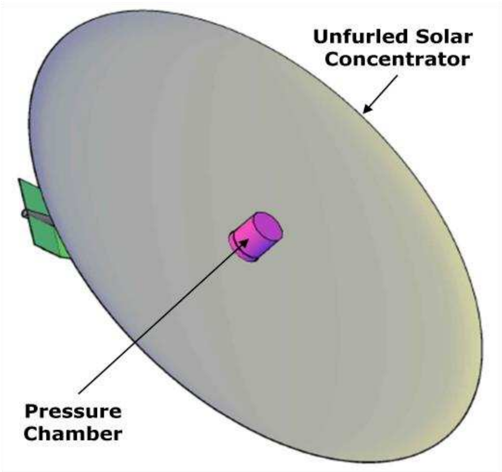
Also the use of Fresnels Concentrators is advised as the cost of multijunction cells is greater than that of normal single junction cells. So the use of Fresnels concentrators allows the increase of the effective light collection area without increasing the physical area of the panel. The technology for multijunction cells and Fresnels concentrators have been flight proven onboard the Deep Space 1 mission, where the panels were used to power the DS1 Ion drive for over 20 months.

Assuming the design specifications of the emcore ZTJ photovoltaic cell, and the ISS Solar Array Wing, the following calculations can be made:
PV Blanket width: 3m
PV Blanket width (extended): 34m
Thus total collecting area: 34*3 = 102m^2.
Total collection area (12 blankets) = 12*102m^2 = 1224m^2
Assumption of Cell Panel Conversion: 200W/m^2.
Thus the total power generated by the 6 panels (each containing 2 blankets) = 244.8kW
This amount of power is sufficient for the operation of the Advanced Xenon engine.

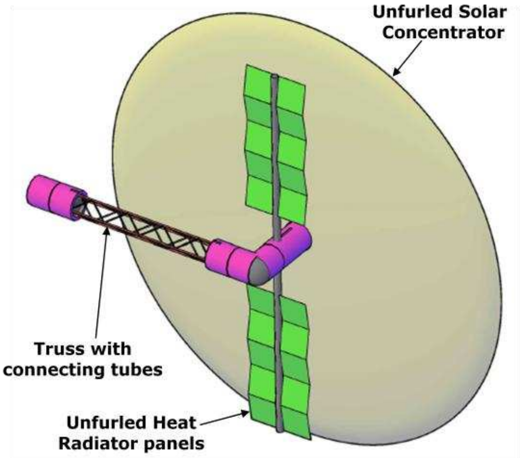
There will be 6 solar wing arrays attached to the propulsion unit, that will supply the same with sufficient power to operate the advanced Xenon ion engine.

7.2 Solar Thermal Dynamic Concentrators

The use of Solar thermal concentrator is three fold in the current mission. Essentially the same as a solar concentrator heater on the surface of the Earth, the use of this method is proposed as an alternative to the Solar photovoltaic method. The designs that do exist, use a Stirling engine and turbine components for the generation of power. Because these components will be bulky for the operation in space, a more affordable configuration is proposed for use on this mission. The conceptual diagram for the unfurled solar concentrator is shown in the following figures.



The unfurled solar concentrator showing the pressure chamber.



The rear of the solar concentrator showing the heat radiators.

There will be two solar concentrators on diametrically opposite points of the habitation module. the following picture shows the folded solar concentrator before unfurling of its components , but with the mast extended.

The solar concentrator has three objectives, out of which the first one is primary and the rest two are secondary.
Primary objective:

- Heat the Carbon Dioxide pumped into the pressure chamber and provide the hot junction for the N-type thermopiles for power generation.

Secondary objectives:

- Preheat the carbon dioxide extracted from the air filtration to reduce electrical requirement for the cracking plant to convert the carbon dioxide into methane.
- The two Solar concentrators on either side of the habitation module operates as a stereoscopic sun sensors acting as a backup for Guidance and Navigation.

7.3 Radiation Shielding and Energy Harvesting mechanism

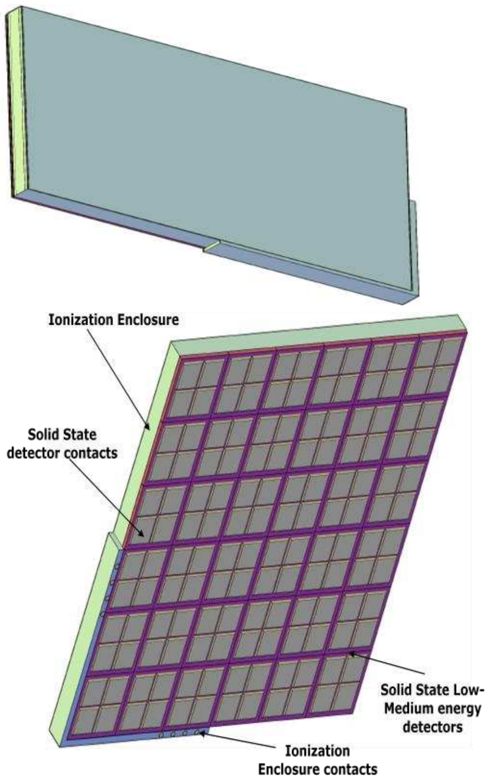
The interplanetary medium consists of several types of harmful radiation that can cause severe damage to human tissue and DNA leading to Cancer. The dosage accumulated by the Curiosity rover on its way to Mars has indicated a sufficient dosage of radiation that accumulates to over 100mSv for a 180 trip to Mars. The accumulation rate was for 1mSv per day due to Galactic Cosmic Radiation (GCR), 5% of which comprised the Solar radiation due to reduced activity at the time of launch. During enhanced solar activity periods this percentage may increase along with the total dosage, thereby increasing the risk due to radiation.

The current space probes outbound of the solar system (e.g. Voyager, Pioneer, New Horizons) as well as the interplanetary probes (e.g. Mars missions rovers, Cassini etc.) all contain electronics which are hardened against radiation. Out semiconductor process technology allows us to develop digital circuits with 28µm processes. But for the purpose of radiation hardening, a more coarser process technology has to be used. But the more bulk material, in the device, the greater is the amount of current required for operation. Hence it is required to optimize the process value and the amount of current required. The effect of radiation on the electronics can be any one of the following or both simultaneously:
(a) Lattice Displacement / Cracking - caused by high energy particle radiation - neutron, gamma, alpha, protons, heavy ions.

They cause a discontinuity in the periodic structure of the crystal lattice.

(b) Ionization effects - caused by low energy charged particles, X-rays, UV rays, which cause a cascade of ion electron pairs, that give rise to soft errors.

For our purpose, we require the Ionization effect to take place, but the Lattice cracking is detrimental to our cause.



The above two diagrams show the radiation harvesting tiles in its assembled form.

The following diagram shows the radiation harvesting tiles in its exploded form.

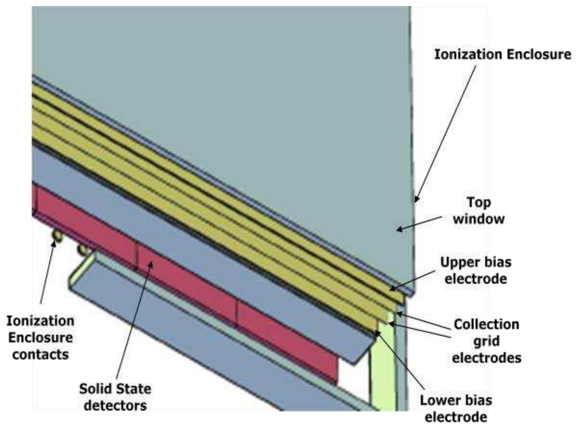


Figure 8 The exploded diagram showing the internal construction of the radiation harvesting tiles.

Ionization chambers are well known to detect particle and high energy radiations (Ionization chambers, Proportional counters and Geiger-Muller counters). There are also low energy solid state radiation detectors, where the current is generated by the resulting of the ion pair formation.

The following diagram shows a tile for the radiation harvesting system.

The upper and the lower bias electrodes are kept at a potential of 200kV. The collection grid electrodes are respectively kept at a potential difference of 50kV and 100kV respectively. The principle of operation is the following:

(a) High energy particle and EM radiation strike the gas molecules in the ionization chamber causing ionization. Neutralization of the ions takes place at the bias electrodes themselves and the collection grids.

(b) For very high energy radiation, a significant amount of energy is lost in the ionization chamber, and the particles now of a lower energy value, impinge on the solid state detectors. The detectors themselves are power photodiodes, thereby converting the cascade from the radiation into useful energy. The lower energy of the ions ensures that the lattice cracking events within the detectors are reduced to a minimum, thereby increasing their operational life, preferably for the duration of the mission.

The efficiency of this system would severely depend upon the components used.

For generating the 200kV required for the ionization chamber and the minimum amount of reverse bias voltage required for the solid state detectors, a small bleed power is used from the primary source - preferably coupled to the solar concentrators and the alternative battery. If a 20V bleed is utilized, then by using a voltage multiplier to up the voltage to 200kV. Since a discharge through the gas is an unwanted phenomenon, hence very little current is to be used - thereby reducing the power draw of the system.

8 Electrical Energy Distribution Subsystem

The primary power from the three sources is coupled into three separate DC-DC converters. A 20V bleed power is used to power a pulsed oil cooled voltage multiplier to power the radiation shielding tiles around the entire spacecraft. The power from the Photovoltaic arrays is used primarily for the ion engine

The power priority by default is the following:

(1) Life Support
(2) Radiation Shielding
(3) Engines
(4) Communication
(5) Guidance & Navigation
(6) Science Payloads

8.1 Solar PV–Battery

One of the most valuable breakthroughs in the space industry was probably the photovoltaic (PV) cell used to convert sunlight into electricity for Earth orbiting satellites. Today, it is the most widely used energy conversion technology in the industry that has fueled the information revolution using high-power communications satellites. Power requirements in tens of watts to several kilowatts over a life ranging from a few months to 15 to 20 years can be met with an array of photovoltaic cells. Satellites requiring continuous load power even during an eclipse must use a rechargeable battery along with the PV array. The battery is charged during sunlight and discharged to power the load during an eclipse. A power regulator and control circuits are used as required for the mission. The general layout of the PV–battery power system is shown in Figure. All components other than the solar array are generally located inside the satellite body. The orientations of the core body and the solar array are maintained relative to the sun and the Earth. The core body is normally maintained in a near constant orientation relative to the Earth, while the α drive and the β gimbals orient the solar array to the sun. The α drive rotates ±360 once per orbit as the satellite revolves around the Earth. The gimbals β rotate ±β to compensate for the variation in the solar angle and also to prevent array shadowing if applicable. Not all satellites have β gimbals, but almost all using the solar energy for power generation have an α drive.

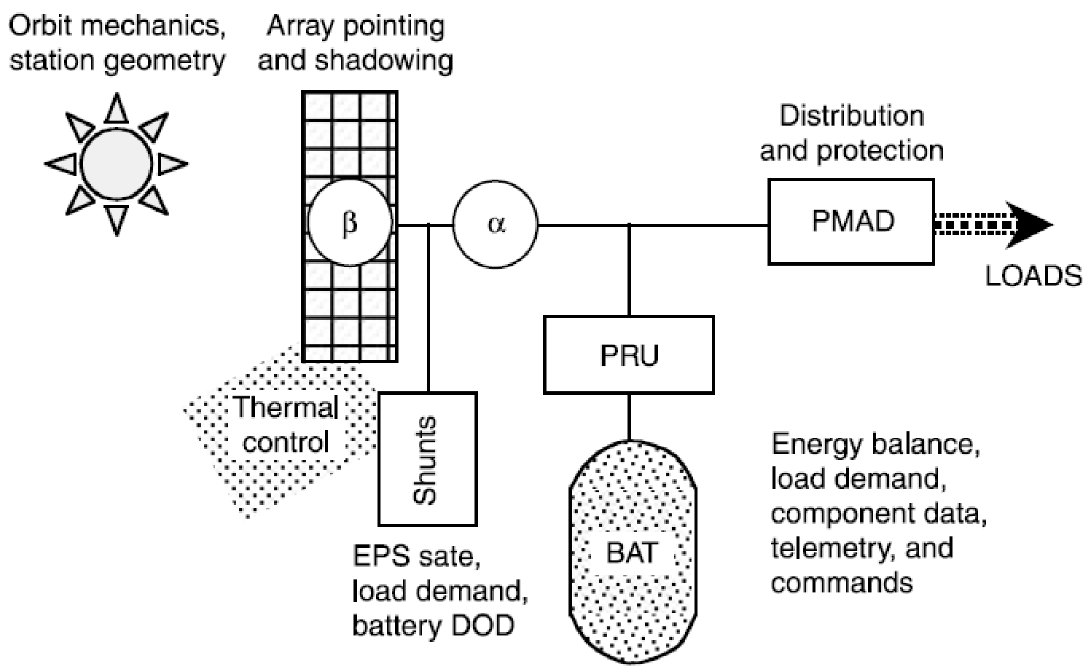


Figure 9 Solar photovoltaic–battery power system configuration.

The most common form of α drive is a slip ring assembly with a solar array drive in 3-axis stabilized satellites, and a rotary power transfer assembly in gyrostats. Angular errors induced by the structural distortions are often compensated by the and/or β drive settings. The seasonal variations of the α angle and the eclipse duration over 1 year for the International Space Station in 400-km (220-n.m.) altitude and 51.6 inclination orbit are shown in Figure. For a given system design, the power available to the load varies over the year due to seasonal variation in the angle. At high when the eclipse duration is zero, the load capability of the electrical power system would be the greatest, as no battery charge power is required. For the ISS, there would be no eclipse at all for _ > 71_, making the orbit sun-synchronous.

The PV cell has been a building block of space power systems since the beginning. The cell is a diode-type junction of two crystalline semiconductors, which generates electricity under sunlight. Its performance at the beginning of life (BOL) is characterized by the output voltage and current at its terminals as shown by the heavy line in Figure 3.7. The two extreme points on this curve, namely the open circuit voltage, Voc, and the short circuit current, Isc, are often used as the performance indicators. The maximum power a cell can generate is the product of Voc, Isc, and a factor that is approximately constant for a given junction. The I–V characteristic of the PV cell degrades as shown by thin lines with the increasing fluence of charged particles on the solar array in the space environment. Such degradation results in decreasing power generation with time. With the combination of seasonal variations of _ angle and yearly degradation of charged particles, the power generation of the solar array over the mission life varies as shown in Figure

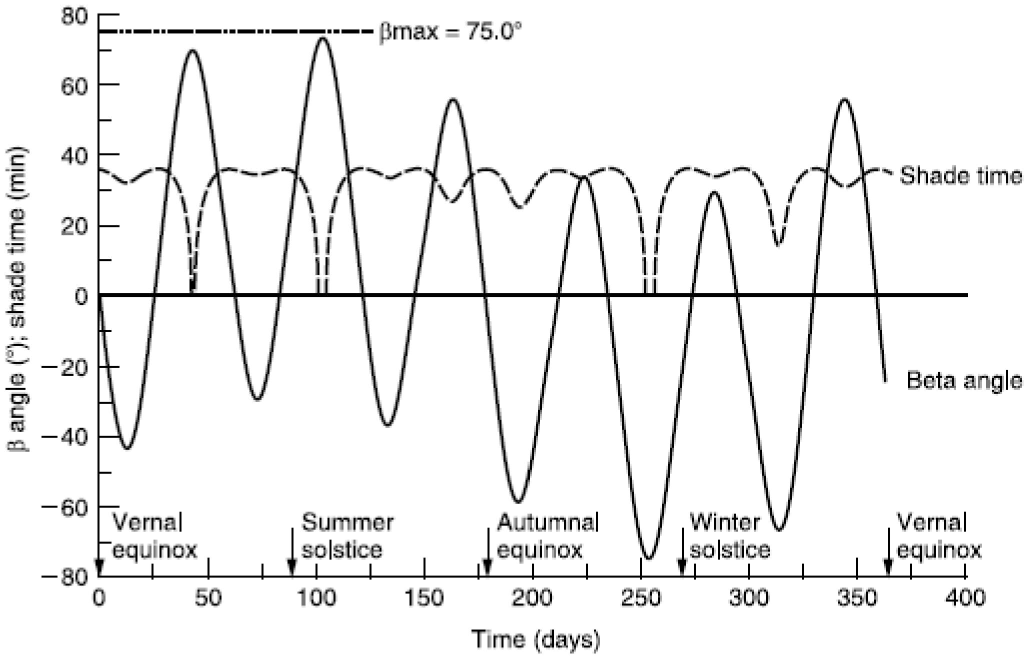


Figure 10 Beta angle and eclipse duration variation with season for the International Space Station. (Source: NASA Glenn SPACE Team/J. Hojnicki.)

9. Thermal Protection Systems

The Space Shuttle thermal protection system (TPS) is the barrier that protected the Space Shuttle Orbiter during the searing 1,650 °C (3,000 °F) heat of atmospheric reentry. A secondary goal was to protect from the heat and cold of space while on orbit. The orbiter's aluminum structure could not withstand temperatures over 175 °C (347 °F) without structural failure. Aerodynamic heating during reentry would push the temperature well above this level in areas, so an effective insulator was needed.

Metallic thermal protection systems (TPS) are being developed to help meet the ambitious goals of future reusable launch vehicles. Recent metallic TPS development
efforts at NASA Langley Research Center are described. Foil-gage metallic honeycomb coupons, representative of the outer surface of metallic TPS were subjected to low speed impact, hypervelocity impact, rain erosion, and subsequent
arcjet exposure. TPS panels were subjected to thermal vacuum, acoustic, and hot gas flow testing. Results of the coupon and panel tests are presented. Experimental and analytical tools are being developed to characterize and improve internal insulations. Masses of metallic TPS and advanced ceramic tile and blanket TPS concepts are
compared for a wide range of parameters

9.1 Metallic TPS concepts

Metallic TPS use a fundamentally different design approach than ceramic tile and blanket concepts. Ceramic tile and blanket concepts require materials that act as a thermal insulator and also perform the structural functions of maintaining the TPS shape and resisting inertial and aerodynamic loads. Metallic TPS concepts seek to decouple the thermal and structural functions by providing a metallic shell to encapsulate internal insulation, maintain panel shape and support mechanical loads. This decoupling allows the use of structurally efficient materials and configurations as well as thermally efficient internal insulations. Of course, the functions cannot be totally decoupled. The structural connections between the outer surface and substructure must be minimized to reduce heat shorts, and the internal insulation must still resist inertial and acoustic loads (perhaps attenuated by the metallic shell). However, this approach opens up a wide range of possible TPS configurations.
Current metallic TPS concepts use a foil-gage, superalloy honeycomb sandwich to form the hot outer surface. Two different configurations are being pursued. NASA LaRC has been studying a superalloy honeycomb sandwich (SA/HC) TPS consisting of lightweight fibrous insulation encapsulated between two honeycomb sandwich panels . The panels are designed to be mechanically attached directly to a smooth, continuous substructure. Each panel is vented to local pressure so that aerodynamic pressure loads are carried by the substructure rather than the outer honeycomb sandwich of the TPS. The outer surface is comprised of a foil-gage Inconel 617 honeycomb sandwich and the inner surface is a titanium honeycomb sandwich with part of one facesheet and core removed to save weight. Beaded, foil-gage, Inconel 617 sheets form the sides of the panel to complete the encapsulation of the insulation. The perimeter of the panel rests on a RTV (room temperature vulcanizing adhesive) coated Nomex felt pad that prevents hot gas flow beneath the panels, provides preload to the mechanical fasteners, and helps damp out panel vibrations.

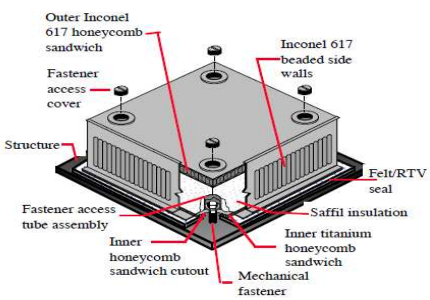


Figure 11 Prepackaged Superalloy Honeycomb TPS Panel.

A prepackaged superalloy honeycomb TPS concept was incrementally improved from a previous design and evaluated for RLV requirements. The superalloy honeycomb TPS concept, illustrated in figure 3, consists of a foil-gage metallic box encapsulating a fibrous insulation. The outer surface of the metallic box is comprised of a honeycomb sandwich with 0.005 in.-thick facesheets and a 0.0015 inch-thick, 3/16 inch cell honeycomb core. The 0.003 inch-thick side walls are beaded to help alleviate thermal stresses and to resist buckling when carrying compressive loads. Both the outer honeycomb sandwich and the sides are made from Inconel 617, a nickel-based superalloy which enables the TPS to operate at a maximum temperature between 1800˚F and 1900˚F -- with limited temperature excursions up to 2000˚F. On two adjacent edges of the panel, the outer facesheet and a flange from 4 the beaded side extend approximately 0.3 inches to form a gap cover. This gap cover is
designed to cover the gap with an adjacent, downstream panel to inhibit hot gas flow in the panel-to-panel gaps. For the 1-foot-square panels, a gap just over 0.2 inches is required to accommodate thermal expansion of the outer surface of the panel. Structural deformations may also play a role in determining the required panel-to-panel gap. A corresponding under hanging lip on the opposite two edges of the panel serve to close off the bottom of the panel-to-panel gap to contain any hot gas flowing in the gap. The inner surface of the metallic box is made of a titanium alloy, Ti 6Al-4V. In the previous design, the lower surface consisted of titanium honeycomb sandwich, with 0.006 inch-thick facesheets and a 0.0015 inch-thick, 3/16 inch cell honeycomb core. Away from the edges of the panel, the facesheets of the sandwich were chem-milled down to 0.003 inch thick. In the current concept the lower chem-milled facesheet and the associated honeycomb core were removed to reduce mass, leaving a 0.003 inch-thick foil on the interior of the lower surface to encapsulate the insulation framed by a section of titanium honeycomb sandwich which stabilizes the lower edges of the sides. A hole, covered by 400 mesh screen to keep out liquid water, is provided in the lower surface of the panel to vent the interior of the panel to the ambient pressure and thus limit the pressure difference supported by the metallic box.

9.2 How Thermal Protection Systems Work

To protect against the heat of friction, engineers use special insulating blankets, foams, and tiles on the skin of the spacecraft. The heat shield or thermal protection system (TPS), which protects against the heat from the engine exhaust plumes, is a more local system that is installed near the throat of the engine nozzles in the base of the vehicle.

Different methods can be utilized to enable a TPS to keep heat from reaching the inside of the spacecraft. One method is to use a covering material that will absorb the heat and radiate it back into space, away from the spacecraft. All materials radiate heat when they get hot. One can feel this whenever one put their hands near something hot like a radiator, a hot stove, or the coals of a campfire. However, only certain materials can radiate heat so efficiently that the heat does not build up within the material and pass it into the spacecraft or possibly melt the body of the spacecraft. Another way a TPS works is to let small bits of itself actually burn and fall away from the spacecraft. These materials neither absorb nor radiate much heat, so when the surface becomes very hot, the material starts to burn and erode. The term that describes this process of material being eroded by heat is ablation.

Keeping the Thermal Protection Lightweight

A launch vehicle's engines can lift a certain amount of weight into orbit. That weight is divided between two parts: the weight of the vehicle itself (including the fuel) and the weight of the passengers and the payload. The more the structure of the vehicle weighs the fewer passengers and smaller payload it can carry (for a particular set of engines).Designers try to keep all the parts of the vehicle, including the thermal protection system, as light as possible so that more of the weight can be used for passengers and payload.

Radiant Thermal Protection System on the Space Shuttle A heavy thermal protection system that came off during re-entry would not work. The Spacecraft's nose cone and the front edges of its wings heat up the most during re-entry. When the Spacecraft is at its hottest, temperatures on these surfaces reach as high as 3,000°F (1,649°C). A product called RCC protects the orbiter's nose and wing leading edges from the highest temperatures.

RCC is a combination of materials called a composite. To make RCC, graphite cloth is saturated with a special resin. Next, layers of the cloth are combined and allowed to harden. Finally they are heated to a very high temperature to convert the resin into carbon. Most of the windward (toward the air flow) surfaces and the base region of the orbiter are protected from heat by silica fiber tiles. There are two kinds of tiles. The high temperature tiles protect areas where temperatures reach up to 2,300°F (1,260°C). These tiles have a black surface coating. The low temperature tiles protect areas where temperatures stay below 1,200°F (650°C). These tiles have a white surface coating. There are approximately 24,300 tiles on the outside of each orbiter. The tiles dissipate heat so quickly that you could hold a tile by its corners with your bare hand only seconds after taking it out of a 2,300°F (1,260°C) oven even while the center of the tile still glows red with heat Some of the leeward (away from the air flow) upper surfaces on the orbiter are protected by flexible insulation blankets. There are 2,300 flexible insulation blankets on the outside of each orbiter. These blankets look like thick quilts.They are made of silica felt between two layers of glass cloth sewn together with silica thread.The blankets are more durable and cost less to make and install than the tiles. The blankets protect areas where temperatures stay below 1,200°F (650°C).The tiles and insulation blankets are bonded to the orbiter with room-temperature vulcanizing (RTV) adhesive. The adhesive will withstand temperatures as high as 550°F (288°C), and as low as -250°F (-157°C) without losing its bond strength.

10.Active Radiation Shielding For Space Exploration Missions

Space radiation environment represents a serious challenge for long duration mission. On LEO the shadow of the Earth and the effect of the magnetosphere, reduces by a factor 4/5 the dose absorbed by astronauts. During a long duration exploration mission, the dose could easily reach and exceed the current dose yearly limits.

For this reason, during at last four decades, means to actively shield the Galactic Cosmic Ray (GCR) component have been considered in particular using superconducting magnets creating a toroidal field around the habitable module. These studies indicated that magnetic shielding with up to a factor of 10 of reduction of the GCR dose could, in principle, be developed.

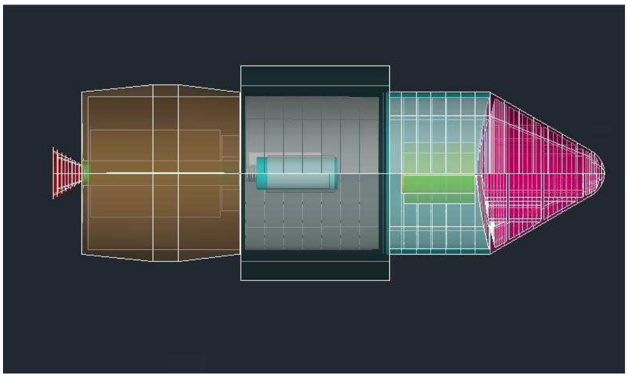


Figure 12 VN Spacecraft with Active radiation shield

Europe has a significant amount of experience in this area thanks previous human missions in LEO (Spacelab, MIR and ISS), related programs (e.g. HUMEX, AMS-02) and Topical Teams activities: this background represent a solid starting point for further progress in this area.

Due to recent and significant progress in superconducting magnet technology both in ground laboratory (ITS – Intermediate Temperature Superconductors - and HTS - High Temperature Superconductors) and on the preparation of space experiments (development of the space qualified AMS-02 superconducting magnet), it is interesting to re-evaluate active shielding concepts as potentially viable solutions to crew protection from exposure to high energy cosmic radiation.

Past active radiation shielding concepts yielded architectures that are significantly massive and too costly to be launched and assembled in space. This is largely due to the magnet size and field strength required to shield Galactic Cosmic Radiation (GCR) and Solar Proton Events (SPE) for meaningful level of crew protection from radiation in space.

Since then, state-of-the-art superconducting magnet technology has made significant progress in performance including higher temperature superconductivity (ITS and HTS) and new mechanical solutions better suited to deal with the Lorentz forces created by the strong magnetic fields. In addition, ten years of design, research and development, construction and testing of the AMS-02 magnet, the only space qualified superconducting magnet built so

Ionizing radiation in space

In space the main contribution to the absorbed dose of ionizing radiation are

Galactic Cosmic Rays (GCR)

Exposure to *GCR* could pose a serious hazard for long-duration space missions. *GCR* radia-tion consists of particles of charge from hydrogen to uranium arriving from outside the heliosphere. These particles range in energy from *~10 MeV n–1* to *~1012 MeV n–1*, with fluence - rate peaks around *300* to *700 MeV n–1*. Because of the vast energy range, it is difficult to provide

far, provides an heritage of European based experience which motivates further developments of this technology for space applications.

Radiation Protection System Roadmap, identifying Ten Critical Technologies needed to develop within the next ten years a realistic DH active magnetic shield, identifying the corresponding Technology Tree as well as the technological R&D and development of demonstrators which are needed over a period of ten years to bring these technologies from the current status to the TRL needed to test such a system in space

The ten critical technologies which have been identified are:

#1 High performances *ITS* and *HTS* cables (*MgB2*, *YBCCO*)
#2 *Double Helix* coil
#3 Cryogenic stable, light mechanics
#4 Gas based recirculating cryogenic systems
#5 Cryocoolers operating at low temperature
#6 Magnetic field flux charging devices
#7 Quench protection for *ITS* and *HTC* coils
#8 Large cryogenic cases for space operation
#9 Superinsulation, Radiation Shielding, Heat Removal
#10 Deployable *SC* Coils

1) Solar Particle Events *(SPE)*

2) Galactic Cosmic Rays *(GCR)*

adequate shielding, thus these particles provide a steady source of low dose-rate radiation.

Integrations of energy spectra show that *~75 %* of the particles have energies below *~3 GeV n–1*. Under modest aluminum shielding, nearly *75 %* of the dose equivalent is due to particles with energies *<2 GeV n–1*. Thus, the most important energy range for risk estimation is from particles with energies below ~2 *GeV n–1*, and nearly all of the risk is due to particles with energies <10 *GeV n–1*. The local interstellar energy spectrum (outside the heliosphere) is a constant, but inside the heli-osphere the spectrum and fluence of particles below ~10 *GeV n–1* is modified by solar activity. The assessment of radiation risk requires a detailed knowledge of the composition and energy spectra of *GCR* in interplanetary space, and their spatial and temporal variation

In the case of deep space interplanetary mission, neglecting the contribution of *SPE*, different *GCR* species contribute to the absorbed radiations with fractions of the total dose depending on the electric charge, due to the energy deposition mechanism based on ionization, which follows a *Z2* law. Due to that reason, *Fe* nuclei, although much less abundant than protons, provide the most important contribution to the radiation dose of all *GCR* nuclear species (*Figure 1.3*).

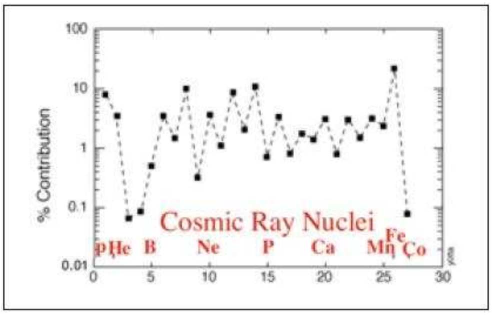


Figure 13 Dose contribution of different CR species

Considering the effect the *11 years* solar modulation, the fluxes of *GCR* are about *40-60%* low-er during solar maximum (*Figure 1.4*): the corresponding doses are *60%* lower during this part of the solar cycle. This is a major difference in dose, which could be used as factor when planning for an exploratory mission.

Doses on exploration missions

For transits to Mars the main concerns are exposures from large SPEs and chronic exposures from both SPEs and the background GCR environment. Since transit times of approximately six months are thought to be necessary, effective doses in excess of 1.100 mSv /y (>110 rem/y) in deep space have been estimated from the GCR environment (see also Table 1.1). Much of this effective dose comes from high-LET components of the spectrum, such as high-energy heavy ions (the so-called HZE particles). Because of weight reasons, typical shielding thicknesses for interplanetary spacecraft are likely not to exceed ~ 8 cm of aluminum (~20 g cm–2) or other structural materials. Doses from large SPEs, mainly from energetic protons with energies as large as several hundreds of mega electron volts and higher, are likely to be well below any acute radiation syndrome response levels for spacecraft with ~15-20 g cm–2 of shielding.

For operations on the surface of Mars, the main sources of concern are chronic exposures to SPEs and the GCR environment. Acute exposures to SPE protons are unlikely because the overly-ing atmosphere of Mars (~16 to 20 g cm–2 carbon dioxide) provides substantial shielding for all sur-face operations, except those that might take place at high mountainous altitudes. The overlying at-mosphere on Mars will also provide some shielding against incident GCR particles. Especially im-portant will be secondary neutrons, which come from nuclear fragmentation interactions between the incident protons and heavy ions and the atmosphere, and from albedo neutrons emanating from the Martian soil(12). These neutron energies range from thermal up to hundreds of mega electron volts or more. Then, the GCR dose on Mars is expected to be between 100 and 200 mSv/y (10-20 rem/y), depending on the location

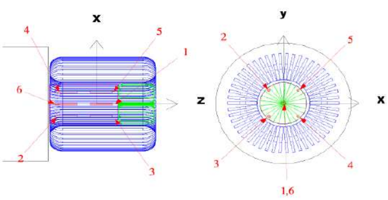


Figure 14 The location of the water cylinders in the habitat

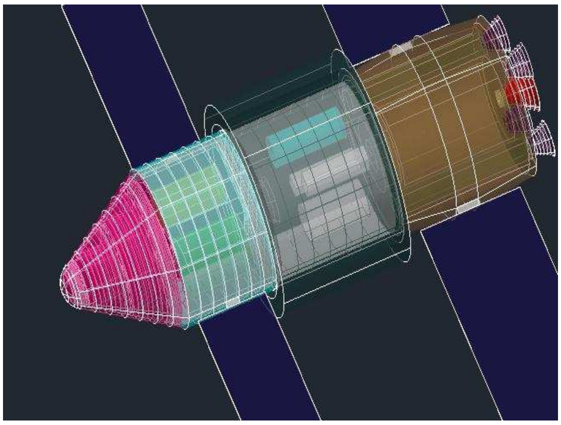


Figure 15. VN Spacecraft with Active radiation shield location

On average astronauts and cosmonauts on *ISS* receive *0,6 mSv d–1* ***(229 mSv y-1)***, with *~75 %* coming from *GCR* and *25 %* coming from protons encountered in passages through the *South Atlantic Anomaly* (*SAA*) region of the Van Allen belts: we recall that a long duration stay on the *ISS* typically does not exceed 6 months. About a factor *5,5* exists from *ISS* to deep space (***1.100 mSv y-1***) where no protection from the magnetosphere or planetary shadow exists: this factor reduces to *3,7* namely *2,2 mSv d–1* (**740 *mSv y-1***) inside a Mars habitable module with *1,5 cm Al* thick walls (*4 g cm-2*).

10.1 Novel Materials Concepts

Carbon Nano-Materials

• Confirmed storage of H up to 6 percent mass fraction1 and reports of up to 20 percent.
• Large and active research base for H storage and materials applications.
• Dual use as shielding and structure/H storage a possibility.
Recommendation: Recommend continued research in this area and liaison with Department
of Energy (DOE) studies.

Metal Hydrides

• Various metal hydrides contain 7–18 percent H.
• LiH has been fabricated for space reactor shielding.
• LiH is competitive with CH2 in shielding cosmic rays.
• LiBH4 contains largest mass fraction of H (18 percent).
• Reactive to various degrees with air and water.
• DOE is studying hydrides for H storage.
Recommendation: Recommend studies of fabrication, encapsulation for hazard abatement,
and liaison with DOE studies on these materials. Recommend assessment of relative shielding
effectiveness using a code such as HZETRN.

Palladium Alloys for Hydrogen Storage

• Higher volumetric density for H.
• Mass fraction of H; ≈4 percent reported.
• High average atomic mass; concern about neutron production.
• May have dual-use applications, particularly where volumetric considerations are important.
Recommendation: Continue present studies and evaluate shielding effectiveness. Recommend
assessment of relative shielding effectiveness using a code such as HZETRN.
1For reference, polyethylene is 14 percent hydrogen by weight.

Polyethylene

• Polyethylene is best "standard or nonnovel" material, except for H, since it contains 14 percent mass
fraction of H and carbon preferentially fragments into 3xHe rather than neutrons.
• In calculations using HZETRN, borated polyethylene is a slightly worse shield than pure polyethylene
because B releases neutrons in interactions as well as absorbing them.
Recommendation: Investigation of possibility of laminates, etc., with pure polyethylene.
Reevaluate borated polyethylene with future improved shielding codes for thicker shields.

Quasi-Crystals

• Absorbed H: 1 to 2.5 percent mass fraction.
• High atomic mass absorbers.
Recommendation: Not competitive with other materials considered here as radiation shield;

not recommended for further study.

Solid Hydrogen

• Has been studied for propulsion (slush H).
• Not a rigid material, and density slightly less than liquid.
• Costly.

Recommendation: No apparent advantages over liquid H2 for shielding; not recommended for study.

11. Scientific Payload

There has been several experiments sent to the red planet regarding the study of the geomorphological structure and the evolution of the planet. But as the intention is to settle on the red planet, it is necessary to conduct experiments on the habitability in the Martian Environment. Some experiments regarding the weather and radiation have already been performed and are also underway for this purpose. But we are limited by the amount of mass we can land on an extraterrestrial surface, which is usually governed by the amount of payload we can lift into orbit. But building a habitation module on Mars for human survival requires more than just radiation and weather sampling. We need oxygen, a credible atmospheric pressure and several resources to survive. We also need to know the surface substructure and radiation levels to ensure the safety of the crew.

In-situ resource utilization have always been the motto for Mars habitation exploration, as the time period for each journey is severely limited by the propulsion techniques we currently employ. Hence we are also required to validate several key technologies to ensure the survival of an entire ecosystem, to preserve human life.

In this context the following experiments are proposed for both scientific benefits and for generating credible data for future manned landing missions.

(1) Inflatable Habitats (Technology proof to fly to ISS between 2015 to 2017 as BEAM):

Inflatable Habitats provide for a greater living / operating volume for a human crew at lower launch weight and cost. It has progressed from the conceptual design phase, and is due to fly to the International Space Station as a module known as the Bigelow Expandable Activity Module in 2015.

(2) Sample Return:

Although many a probes have been sent to the surface of the red planet, none of the probes have been able to return a sample from the red planet, due to technical reasons. This mission provides a clear and present opportunity to perform a sample return from the surface of the planet. But since this mission is a flyby, the time period for orbital capture of the samples is not present. But the crewed vehicle can act as a carrier vehicle by dropping the sample return modules on the surface. The Martian gravity is 1/3 that of Earth Sea Level, hence a small rocket would be able to help it achieve orbit and put the return capsule on a course for Earth. The module will blast off from Mars surface using two small solid fuel rocket to reach orbit and a second stage liquid engine will provide enough thrust to break Mars SOI and for Earth capture. [Genesis]

In-Situ resource utilization has been considered as one of the most important part of any manned Mars mission because of the time periods involved for travel and the technological limitations that are present in the current state. Given below are some of the experiments that should be conducted as separate modules to test for the In-Situ resource utilization technology:

(3) Oxygen Extraction Experiment:

Oxygen is required for both human survival and also for an oxidizer for the rocket fuel. The OEE module contains two vital components that would work off the solar power. A modified form of the Sabatier reactor on the Habitation module to produce enough O_2 and CH_4 to act as rocket fuel for the sample return mission. It will carry the seed water with it to ensure the conversion. This fuel and oxidizer content would be used to accelerate the rocket or can be vented into space, depending upon the judgment of the mission planners and the required thrust. The experiment will confirm two basic technologies required for the future missions - the ability to implement in-situ resource utilization using robotics for future missions and generating the necessary resources for human habitation and rocket fuel to supply the human base.

(4) Water Extraction Experiment:

Along with Oxygen the second most important resource required is water. It is well known that the polar caps of Mars contain solid water under the surface. The 2nd core drill has an integrated sampling and heating element, that will vaporize the trapped water and then condense it to extract liquid water for base use. this would be another verification of the In-situ resource utilization technology.

(5) Greenhouse (COSPAR norms implemented):

Although it is known very well that Mars is geologically a dead planet, the habitation on Mars surface may encounter un-foreseen circumstances where the Oxygen extracted from the air may be inadequate or the system may break down. In this case a natural method of Oxygen would be preferred using trees. But unfortunately, neither the temperature, nor the soil and nor the atmospheric condition of Mars currently enable plantations, and most importantly the COSPAR rules states that all probes are to be thoroughly sterilized to avoid contamination of the Extraterrestrial environment. the greenhouse system is a small high pressure glass dome, that contains earth soil and gram seeds, but uses Martian carbon dioxide and Martian sunlight to investigate the effects on terrestrial flora. The small seed will have to be packed into the glass dome before the drop of the landing package. In the event of a containment failure and possibility of contamination of the Martian environment, a small propylene container will release its gas content into the glass dome, and the system will be boosted to a high altitude, using a flat solid thruster rocket. an ignition system will the ignite the gas oxygen mixture to ensure the complete burning of all terrestrial contaminants at high temperature, resulting in charred carbon. A small camera will regularly send photographs of the plant for ground control to determine its growth.

(6) Radiation Monitoring Experiment:

As similar with the RAD experiments on other missions, the radiation experiment will determine the daily dose of radiation on the lander to determine the exposure of the habitation module of the incoming radiation.

(7) Mars Seismograph:

Prior to the touchdown of the lander, a small dispersal system, will release MEMS based solar powered accelerometer nodes to cover over a sufficiently large area. On landing, a small cluster of antannaes will identify and determine the position of each sensor node. An ad-hoc communication system is not advised as the computational power needed would require too complex electronics to implement. The lander itself contains 6 slugs in a RCL launcher. As it fires each slug, the resulting pressure wave travel to each sensor node and the data recovered is transmitted back to Earth or the ship for interpretation. The data may help us to develop an understanding of the Martian subsurface in order to plan for future buried Mars Habitats or to develop the foundations for an above ground base so that its effect in Martian dust storms are minimal.

(8) Mineral extraction and refining:

This experiment is not possible to be implemented on the Martian budget because, the power budget of the lander, would be incapable of generating a hermatic vacuum seal. The system will be powered by solar photovoltaic concentrator arrays, powering a broad power spectrum electron or hydrogen beam, that causes secondary ionization of the elements on the target surface. These ions then float towards a negative potential, and is differentiated using a magnetic field. The ions will be neutralized and deposited on piezo wafers, whose frequency of vibration changes with the amount of material deposited on it. This experiment is a SIMS proof concept for Extraction and refining of minerals present in the Martian soil. The unit will be tested on one of Mars moons - either Phobos or Deimos.

Power and Weight budgets for the scientific payloads:

The Inflatable Habitat may or may not be ready for the mission, hence it is optional. The Mineral Extraction and Refining Experiment is to be performed on Phobos or Deimos, thereby making it a separate unit with a solar array generating 150W and the necessary power storage systems. The rest of the experiments are bundled into one lander system having a mass of approximately 2,000 kg. powered by two solar arrays generating approximately 100KW

Experiments	**Mass – estimated**	**Power consumption (W)**	**TRL**
Inflatable Habitats	5,000 kg.	----	9
Sample Return system	600 - 1,000 kg.	50	3 - 9 (separate subsections)
Oxygen Extraction Experiment	37	25	9 (subsections)
Water Extraction Experiment	3 – 4	4	5
Greenhouse	30 kg.	10	2 (plants flown in space - TRL-9)
Radiation Monitoring Experiment	1 - 2 kg.	5	9
Mars Seismograph	0.9 - 1 kg.	0.3 (electronics only)	7
Mineral Extraction and Refining	1000 kg.	~ 150	2

Table 3 Habitability and mass estimation

12. Environmental Control and Life Support System (ECLSS)

Introduction

The Environmental Control and Life Support System (ECLSS) maintains the pressurized habitable environment of the crew modules. Each of the pressurized modules will be equipped with two separate units, with any one unit operating at any given point of time. This allows for redundancy, safety measures, and a credible backup in the event of a failure. The various objectives of the ECLSS can be summarized in the following given below:

(a) Carbon Dioxide Removal
(b) Temperature and Humidity Control
(c) Water Recovery and Management
(d) Maintenance of Cabin Atmosphere and Pressure
(e) Fire Detection and Suppression
(f) Oxygen Generation

A majority of these systems are similar if not the same as those used on the International Space Station. This would ensure a mature level of technology supporting the ECLSS system on the mission as well as ease of construction to meet the 2018 deadline. A few major additions are also made, keeping in mind the nature of the mission and the distance from Earth. On the ISS, several of the products of the ECLSS are vented outboard, and the resupply missions, provide more raw material to keep the system running. This is not an option for the Mars mission, because of the distance and the time of the mission. To ensure a successful habitable environment, a closed system based on the Sabatier reaction has been proposed to sustain the astronauts for the mission duration. An overview of each of these systems in brief is given below.

Carbon Dioxide Removal (CDR)

This will be a small array of regenerable sorbent beds, that will absorb the Carbon Dioxide. During the absorption process the CDRA requires cold, dry air to flow through it, and hence is connected to the temperature and humidity control system. The number of astronauts and the volume of the enclosed habitat module on this mission, does not require large sorbent beds but numerous number of small beds that can be used as both primary and backup, when the need arises. LiOH can be provided for emergency CO_2 removal in the event of a complete failure of all sorbent beds.

Temperature and Humidity Control (THC)

This sub-module is responsible for temperature humidity control of the habitable section. The Temperature is controlled either actively using small Peltier coolers in every module or passively using retractable radiating panels. The humidity control, used to maintain the humidity content of the atmosphere is interfaced with the water recovery system, to facilitate the humidity settings of the crew.

Water Recovery and Management (WRM)

The WRM sub-module recovers the water from various sources and recycles it for further use by the crew. Attempts should be minimize the amount of water loss due to atmosphere leakage. Since the water that is recycled is also required for Oxygen production using the closed Sabatier cycle, the WRM module is interfaced with all other modules of the ECLSS.

Atmosphere and Pressure Maintenance (APM)

The Atmosphere and Pressure maintenance is directly interfaced with the oxygen generator, the nitrogen mixer systems and the Temperature and Humidity control systems. A double hulled construction for the modules to arrest the already minimal loss of atmosphere due to several imperfections during construction.

Fire Detection and Suppression (FDS)

In the event of a fire, the required module is sealed off, and the Carbon Dioxide storage system is expunged to extinguish the fire. Under an extreme condition, and only as a last resort, the module can be depressurized by venting into space. After the fire has been extinguished, the backup sorbent beds in the module can be brought online to remove the excess CO_2. The crew will also be issued with portable oxygen breathing apparatus and fire extinguishing systems to allow them to fight a fire.

Oxygen Generation (OG)

The Oxygen Generator is a closed cycle Sabatier Reactor, an open cycle prototype of which is already in operation onboard the ISS, operating in conjunction with the Solid Fuel Oxygen Generators and the Elektron Oxygen generation systems.

The respiration of the astronauts convert the oxygen in the module into carbon dioxide according to the following equation:

$$O_2 \rightarrow (respiration) \rightarrow CO_2$$

the CO_2 is scrubbed from the atmosphere using the CDR system, and recycled as follows.

$$CO_2 + 4H_2 \rightarrow 2\mathrm{H}_2O + CH_4$$

The electrolysis of water would generate the required oxygen, and the hydrogen required to sustain the reaction.

$$2H_2O \rightarrow (electrolysis) \rightarrow 2H_2 + O_2$$

The Methane can be recycled back to generate the hydrogen under pyrolytic cracking.

$$CH_4 \rightarrow C + 2H_2$$

This transformation can be achieved under high temperature without the presence of oxygen. The following schematic diagram shows the quartz pyrolytic cracking chamber to perform this process:

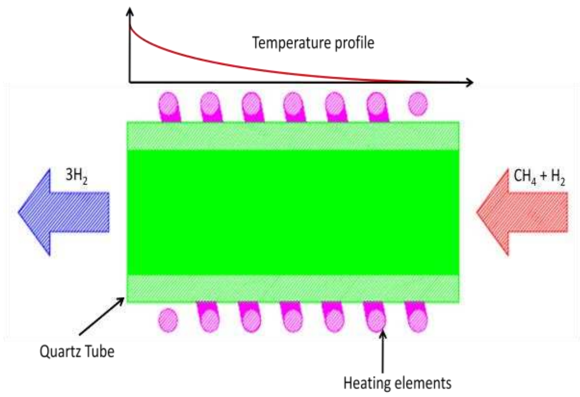


Figure 16 Quartz tube

The Quartz tube is chose so as to withstand the high temperatures, that are prevalent of this process. As the reaction continues, the carbon will get deposited along the walls of the tube.

Operation and cleaning of the Conversion tubes.

As the carbon gets deposited along the length of the quartz tube, the efficiency and the cracking rate of the system will decrease. For this reason, each transformation unit shall be equipped with three such transformation tubes, with anyone tube operating at a certain point of time, for the duration of a certain period. When the conversion efficiency drops below 95%, that certain unit will be turned off and another unit will be brought online. The following protocols need to be observed:

(a) All three units should not remain operationally off for more than 5 minutes, as there may be a chance of contamination of the cabin atmosphere with the methane.

(b) Before a unit can be detached or cleaning, a second unit must be brought online and remain operational at a minimum of 98% efficiency for a minimum of 3 hours.

NASA has already been successful in the development of a small Sabatier system that may be used for the mission at hand.

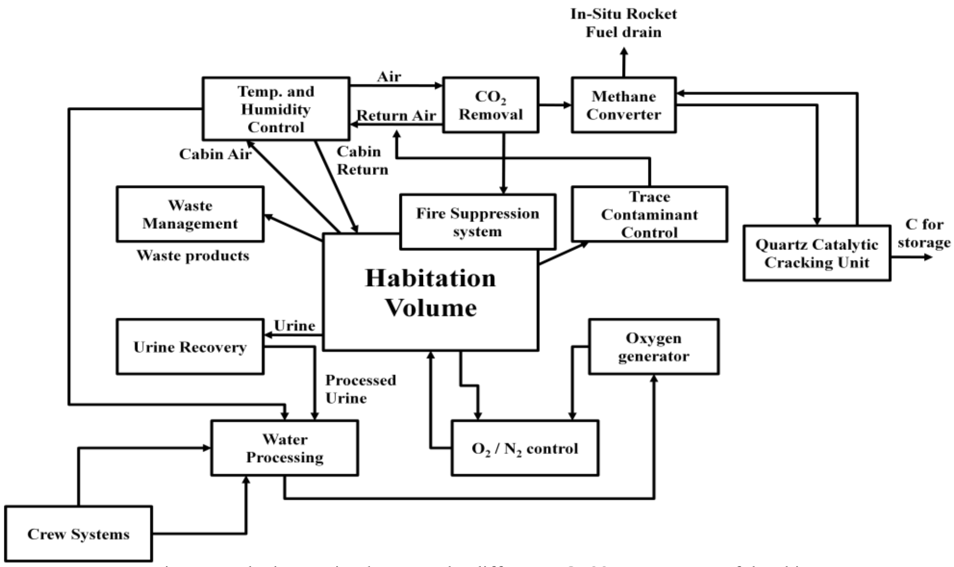
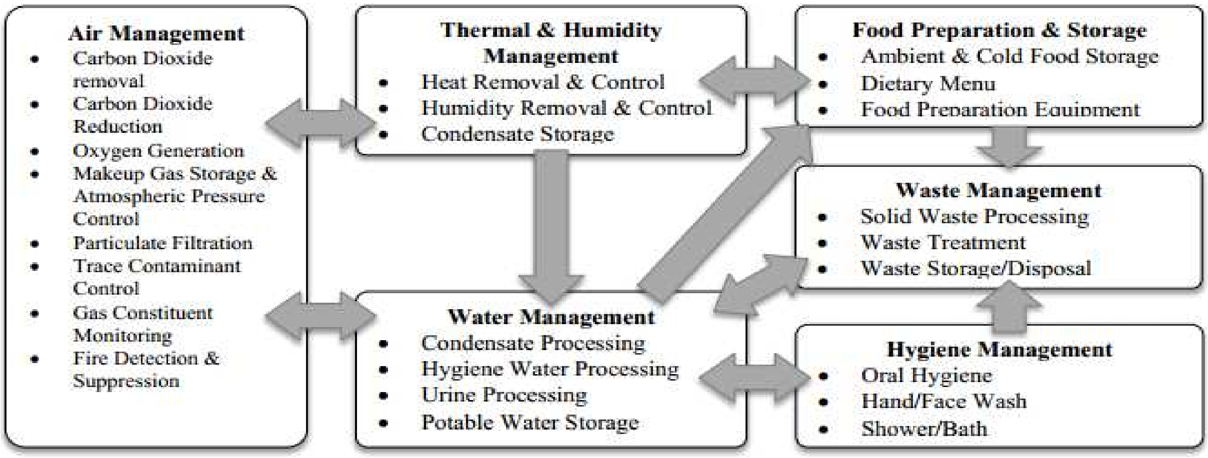


Figure 17 the interaction between the different ECLSS components of the ship.

The cabin atmosphere assumes 101kPA (14.7psi) with a nominal makeup of 78% Nitrogen (N2), 21% Oxygen (O2), and 1% other. Subsystem components and stored gasses are included for makeup gasses for leakage and gas equivalent storage for one cabin repressurization, though no depressurization accommodations are included (e.g. no pressure suits). We reviewed numerous technology options to revitalize the cabin air, and determined that an electrolyzer and Sabatier work in tandem to provide necessary air revitalization needs. The electrolyzer produces O2 and H2 from water. The Sabatier processes CO2, but needs H2 to do so, which it receives from the electrolyzer. The amount of water needed to fully replenish all O2 requirements was included in the water subsystem calculations, and the difference in H2 production and H2 requirements are balanced to determine the stored H2 required for this subsystem.

The water subsystem assumes basic metabolic requirements for drinking, hygiene, and food processing. Water is added to accommodate O2 production. Various configurations include no recycling, recycling condensation from the atmosphere, and recycling water from the biological waste. Hygiene water is a variable that can be used to reduce mass for this subsystem, and water recycling is assumed for reduced launch mass, discussed further in section 8, discussing ECLSS sizing. The SOA technology for condensate recovery utilizes multi-filtration, ion exchange, and catalytic oxidation. Food estimates include packaging and storage factors, as well as a galley and food processing equipment such as heating, preparing, and disposing of waste. While it is probable that all food is consumed, factors to account for food adhesion to the packages are included. The thermal system is based on a redundant loop radiator, and is sized for the ECLSS power as estimated for the different configurations, plus an assumed 2kW of heat coming from air-cooled components such as avionics.

The waste system provides the interface for collecting and storing human biological waste. Technologies that are used to recycle water from the liquid waste are included in the water system as noted above. Personal provisions are limited to items such as clothing and hygiene products. Free volume estimations are based on minimum NASA requirements, and are only used to calculate makeup gasses that are required for the full duration. No consideration of privacy, or separate sleeping quarters was contemplated for this study. The mass and volume associated with free volume needs are not included in the ECLSS mass and volume, as we assume that they will be carried at the overall vehicle level, as is typical.



Long term exposure to microgravity has demonstrated that deconditioning must be counteracted with routine vigorous exercise so we assume the use of resistive exercise equipment that provides full range of motion exercise of all primary muscle groups. Medical equipment and supplies are provided to address emergencies. In addition to spacecraft materials of construction, radiation protection is provided by a water shield made up of water in storage for other subsystems. Other options reviewed were Hydrogen-Impregnated Carbon Nano-fibers, and a Liquid Hydrogen Shell. Further study needs to be done to find creative solutions for radiation protection, including the amount of radiation and the level of risk of a high radiation event deemed to be acceptable. Evaluating these risk factors will include an exploration of crew age, gender, and various exposure types; and will affect future ECLSS trade studies.

Interface	Avg. Single Crew-Member per Day			Total 2CM 500d*	
Overall Body mass		70	kg	140	Kg
Respiratory Quotient		0.869		N/A	
	Input	*Output*			
Air					
Carbon Dioxide Produced		0.998	kg /CM-d	998	Kg
Oxygen Consumed	0.835		kg /CM-d	835	Kg
Water					
Potable Water Consumed	3.909		kg /CM-d	3,909	Kg
Fecal Water		0.091	kg /CM-d	91	Kg
Respiration and Perspiration Water		2.277	kg /CM-d	2,277	Kg
Urine Water		1.886	kg /CM-d	1,886	kg
Metabolically produced Water	0.345		kg /CM-d	345	kg
Food					
Dry Food Consumed	0.617		kg /CM-d	617	kg
Thermal					
N/A					
Waste					
Fecal Solid Waste (dry basis)		0.032	kg /CM-d	32	kg
Perspiration Solid Waste (dry basis)		0.018	kg /CM-d	18	kg
Urine Solid Waste (dry basis)		0.059	kg /CM-d	59	kg

*Does not include packaging and storage containers.

Table 4 Crew metabolic interface values (hanford, 2006, table 3.3.8) [18]

Component	System Total*			
	Mass *(kg)*	Vol† *(m^3)*	Peak *(W)*	Avg *(W)*
Air *(high press storage, mole sieve, Sabatier, electrolysis)*	897	1.7	2,626	1,870
Water *(tanks, multi-filtration, ion exchange, distil, catalytic oxidation)*	2,235	5.1	529	193
Food *(dry packed, storage, water heater)*	1,384	14.0	1,860	39
Thermal‡ *(nominal redundant single loop)*	479	1.0	300	99
Waste *(urine, feces, and vomitus collection, and solid waste stabilization/storage)*	259	0.7	174	7
Human Accommodations *(clothing, hygiene, medical provisions, radiation shelter, etc.)*	347	1.8	-	-
Basic System:	2,432	6.6	5,489	2,208
Consumables:	3,131	17.7	-	-
Total:	**5,545**	**24.2**	**5,489**	**2,208**

*Mass and Power estimates based on ANSI/AIAA G-020-1992, Guide for Estimating and Budgeting Weight and Power Contingencies for Spacecraft Systems
† Volumes are total volume and does not account for packaging factors.
‡ Thermal System assumes that the base spacecraft has thermal control capability for existing systems.

Item	Mass (kg/CM-d)	Total 2CM 500d (kg)*
Feminine Health	0.008	8
Toilet Paper	0.028	28
Gloves	0.007	7
Dry Wipes	0.013	13
Detergent	0.058	58
Disinfectant	0.056	56
Paper	0.077	77
Tape	0.033	33
Clothing	0.486	486
Total (Female)	**0.766**	**766**
Total (Male)	**0.758**	**758**

*Does not include packaging and storage containers.

Table 5 Crew personal provisions (hanford, 2006)[18]

Space Food

Productive, reliable, and safe human space exploration depends on an adequate food system to provide the crew with safe, nutritious, and acceptable foods for up to 5 years with minimal impact to mission resources. The food system is the sole source of nutrition to the crew. A significant loss in nutrition, either through loss of nutrients in the food during processing and storage or inadequate food intake due to low acceptability, variety, or usability, may significantly compromise crew health and performance. Recent research has indicated that the current food system will not meet the nutrition, acceptability, or resource requirements of a long duration mission beyond low-Earth orbit.

There are eight categories of space food:

Rehydratable Food: The water is removed from rehydratable foods to make them easier to store. This process of dehydration (also known as freeze drying) is described in the earlier Gemini section. Water is replaced in the foods before they are eaten. Rehydratable items include beverages as well as food items. Hot cereal such as oatmeal is a rehydratable food.

Thermostabilized Food: Thermostabilized foods are heat processed so they can be stored at room temperature. Most of the fruits and fish (tuna fish) are thermostabilized in cans. The cans open with easy-open pull tabs similar to fruit cups that can be purchased in the local grocery store. Puddings are packaged in plastic cups.

Intermediate Moisture Food: Intermediate moisture foods are preserved by taking some water out of the product while leaving enough in to maintain the soft texture. This way, it can be eaten without any preparation. These foods include dried peaches, pears, apricots, and beef jerky.

Natural Form Food: These foods are ready to eat and are packaged in flexible pouches. Examples include nuts,
granola bars, and cookies.

Irradiated Food: Beef steak and smoked turkey are the only irradiated products being used at this time. These products are cooked and packaged in flexible foil pouches and sterilized by ionizing radiation so they can be kept at room temperature. Other irradiated products are being developed for the ISS.

Frozen Food: These foods are quick frozen to prevent a buildup of large ice crystals. This maintains the original texture of the food and helps it taste fresh. Examples include quiches, casseroles, and chicken pot pie.

Fresh Food: These foods are neither processed nor artificially preserved. Examples include apples and bananas.

Refrigerated Food: These foods require cold or cool temperatures to prevent spoilage. Examples include cream cheese and sour cream.

Nutrition Composition Breakdown

Nutrients	**Daily Dietary Intake**
Protein	0.8 g/kg And ≤ 35% of the total daily energy intake And 2/3 of the amount in the form of animal protein and 1/3 in the form of vegetable protein
Carbohydrate	50-55% of the total daily energy intake
Fat	25-35% of the total daily energy intake
Ω-6 Fatty Acids	14 g
Ω-3 Fatty Acids	1.1 - 1.6 g
Saturated fat	<7% of total calories
Trans fatty acids	<1% of total calories

Cholesterol	< 300 mg/day
Fiber	10-14 grams/4187 kJ
Fluid	≥ 2000 mL
Vitamin A	700-900 μg
Vitamin D	25 μg
Vitamin K	Women: 90 μg Men: 120 μg
Vitamin E	15 mg
Vitamin C	90 mg
Vitamin B12	2.4 μg
Vitamin B6	1.7 mg
Thiamin	Women: 1.1 μmol Men: 1.2 μmol
Riboflavin	1.3 mg
Folate	400 μg
Niacin	16 mg NE
Biotin	30 μg
Pantothenic Acid	30 mg
Calcium	1200 - 2000 mg
Phosphorus	700 mg And ≤ 1.5 x calcium intake
Magnesium	Women: 320 mg Men: 420 mg And ≤ 350 mg from supplements only
Sodium	1500 - 2300 mg
Potassium	4.7 g
Iron	8 - 10 mg
Copper	0.5 - 9 mg

Table 6 Nutrition Composition Breakdown

Current ISS crewmembers receive about 1.8 kg of food plus packaging per person per day. Compared to the Apollo missions a higher percentage of the food is thermostabilized, as a result of crew preference, contributing to the weight increase. Since ISS uses solar panels for a power source, and not fuel cells that produce water as a by-product, there is little mass advantage to using freeze-dried foods. Furthermore, the average number of calories is now based on the actual caloric needs of each crewmember according to body weight and height. This results in an average caloric requirement of 3,000 kcal as opposed to the 2,500 kcal provided to the Apollo crews, and a corresponding food weight increase

Bio-Suit for Crew Members

Existing space suits used hard fiberglass or metal and soft fabric components. Mobility was obtained by pleats that opened as joints bent and rotational bearings. These suits, all derived from the very fist purpose-designed spacesuits of the 1960's, were heavy, bulky, restricted astronaut mobility, and required extensive special training and exhausting joint torque force to work in. A modern Mechanical Counter Pressure (MCP) suit, first studied by NASA in 1971 as the Space Activity Suit, would eliminate these difficulties. The Bio-Suit study did not identify a specific design, but rather identified candidate technologies for the suit layers and the embedded information systems. These included:

- Electric Alloy Mesh Concept, using a seamless Shape Memory Alloy mesh to generate voltage-controlled mechanical counter-pressure. Pressure would be distributed by a viscous thermal-regulating gel layer. The gel layer moderated the high temperature of the

SMA later and protected the body against impacts the skin directly, wicking away perspiration and absorbing body heat.

- Thermal Gel Suit Concept, using "smart" polymer gels which expanded at a threshold temperature to create mechanical counter-pressure. The smart gel was trapped in a quilted layer beneath a stretchless restraint layer. The restraint layer prevented outward expansion of the gel, directing the pressure inwards against the body.
- Electric Gel Suit , using "smart polymer gels which expanded in an electric field to create mechanical counter-pressure. The smart gel was trapped in a quilted layer, between metallized fabric layers, beneath a stretchless restraint layer. Opposite charges applied to the metallized layers produced a small electric field sufficient to stimulated the expanding smart gel.
- Stretch Alloy Band Suit Concept using the super elastic properties of Shape Memory Alloys (SMA) to allow the suit's volume to expand enough for donning. Charge would then applied to the SMA band which pulled together the seam of a uni-directional stretch fabric layer, which was able to stretch longitudinally in order to allow flexion at the joints.
- Electric Alloy Zipper Suits using shape memory alloy strips to aid and control the application of mechanical counter pressure while manually zipping together seams in a uni-directional stretch fabric layer.
- Electric Alloy Remote Zipper Suit concept, as the previous concept, but instead of being zipped manually, tightened all at once by digital controls at the shoulders. This system assured uniformity of mechanical counter-pressure and ease of operation.

The study also looked at various alternatives for thermal control:

- Absorb concept, which would collect perspiration in a removable component within the suit, either a highly absorptive fabric layer similar to long underwear, or desiccant packs at critical locations.
- Vent-to-Atmosphere concept, which controlled perspiration by venting moisture directly to the outside environment. A selective, semi-permeable organic layer closest to the skin allowed perspiration to pass through at a moderate rate. Subsequent layers of the suit, including the mechanical counter-pressure layer, were also semi-permeable. The openings in the membranes were large enough to allow the suit to breath, but small enough to prevent unwanted fluid loss.
- Transport concept, using a layer of tiny tubes to channel perspiration away from the body to a remote collection point. These tubes might be manufactured or perhaps organic such as the aquaporin network in plant membranes. A partial vacuum at the collection end might moved perspiration through the tubes, or perhaps work would be done by tiny piezoelectric pumps powered by energy harvested from body motion.

An advanced possibility was that the suit layers could be sprayed directly on the astronaut's skins prior to EVA. Electrospinlacing, involving charging and projecting of tiny fibers of polymer directly onto the skin, could be used. Melt blowing of liquefied polymer could be used to apply thin elastic layers. Application could be made directly to the skin, or to advanced 3D forms generated by laser scanning. Wearable computers, smart gels and conductive materials could be embedded between polymer layers.

The Bio-Suit is an experimental space activity suit under construction at the Massachusetts Institute of Technology at the direction of professor Dava Newman, with support from the NASA Institute for Advanced Concepts. Similar to the SAS in concept, the BioSuit applies a number of advances in engineering and measurement to produce a dramatically simplified version of the SAS design.

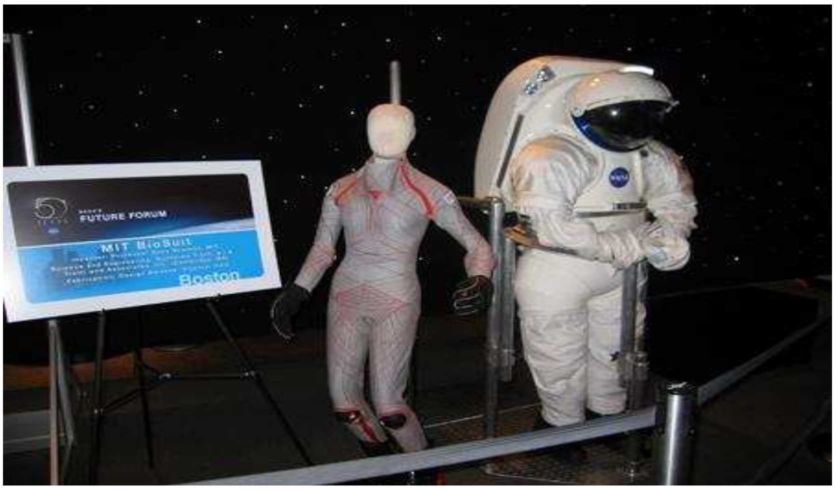


Figure 18 Bio-suit

The primary structure of the BioSuit is built by placing elastic cords along "lines of non-extension", along which the skin does not stretch during most normal movements. Thus, whatever pressure they provide will be constant even as the wearer moves. In this way, they can very accurately control the mechanical counter-pressure the suit applies. The rest of the suit is then built up from spandex lying between the primary pressure cords. The Bio-Suit team has thus far constructed a number of lower leg prototypes using different materials, including nylon-spandex, elastic, and urethane-painted foam. In one experimental design, kevlar fabric was used between cords for areas where the expansion was limited. At least one full-body suit has been constructed for Newman, which she has worn for numerous photo-ops; it is unknown if the entire suit meets the same counter-pressure standards that the lower-leg prototypes were designed for. Each suit has to be custom tailored for the wearer, but the complexity of this task is greatly reduced through the use of whole-body laser scans.

The result is a one-layer version of the SAS; it is lighter than the original and considerably more flexible, allowing much more natural motion and decreasing the energy cost of motion. Current versions of portions of the BioSuit have consistently reached 25 kilopascals (190 mmHg; 3.6 psi), and the team is currently aiming for 30 kilopascals (230 mmHg; 4.4 psi) for a baseline design. As mechanical counter-pressure has proven difficult for small joints such as those in the hands, the BioSuit baseline design uses gas-filled gloves and boots, in addition to a gas-filled helmet.

13. Reentry design

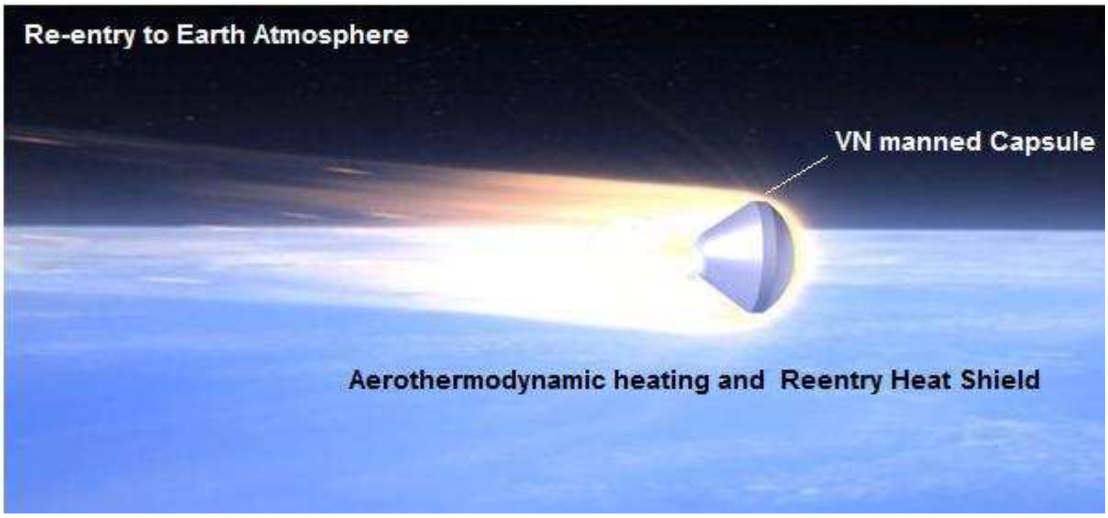


Figure 19 Re-entry of VN Space Capsule

All space-mission planning begins with a set of requirements we must meet to achieve mission objectives. The re-entry phase of a mission is no different. In our design we must delicately balance three, often competing, requirements

- Deceleration
- Heating
- Accuracy of landing or impact

The vehicle's structure and payload limit the maximum deceleration or "g's" it can withstand. (One "g" is the gravitational acceleration at Earth's surface—9.798 m/s^2.) When subjected to enough g's, even steel and aluminum can crumple like paper. Fortunately, the structural g limits for a well-designed vehicle can be quite high, perhaps hundreds of g's. But a fragile human payload would be crushed to death long before reaching that level. Humans can withstand a maximum deceleration of about 12 g's (about 12 times their weight) for only a few minutes at a time. Imagine eleven other people with your same weight all stacked on top of you. You'd be lucky to breathe! Just as a chain is only as strong as its weakest link, the maximum deceleration a vehicle experiences during re-entry must be low enough to prevent damage or injury to the weakest part of the vehicle. But maximum g's aren't the only concern of re-entry designers. Too little deceleration can also cause serious problems. Similar to a rock skipping off a pond, a vehicle that doesn't slow down enough may literally bounce off the atmosphere and back into the cold reaches of space. apart from this limitations of heating also come into picture.

Key Concepts in reentry design

➤ We must balance three competing requirements for re-entry design

- Deceleration
- Heating
- Accuracy

Re entry design

➤ We base the re-entry coordinate system on the

- Origin—vehicle's center of gravity at the beginning of re-entry
- Fundamental plane—vehicle's orbital plane
- Principal direction—down

➤ During re-entry, we can assume

- Re-entry vehicle is a point mass
- Drag is the dominant force—all other forces, including gravity and lift, are insignificant

➤Ballistic coefficient, BC, quantifies an object's mass, drag coefficient, and cross-sectional area and predicts how drag will affect it

- Light, blunt vehicle—low BC—slows down quickly
- Heavy, streamlined vehicle—high BC—doesn't slow down quickly

➤To balance competing requirements, we tackle the problem of re-entry design on two fronts

• Trajectory design—changes to re-entry velocity, V re-entry , and re-entry flight-path angle, γ

• Vehicle design—changes to a vehicle's size and shape (BC) and thermal-protection systems (TPS)

➤We can meet re-entry mission requirements on the trajectory front by changing

• Re-entry velocity, V re-entry

• Re-entry flight-path angle, γ

➤Increasing re-entry velocity increases

• Maximum deceleration, a_{max}

• Maximum heating rate, q_{max}

➤Compared to the drag force, the gravity force on a re-entry vehicle is insignificant

➤Increasing the re-entry flight-path angle, γ , (steeper re-entry) increases

• Maximum deceleration, a_{max}

• Maximum heating rate, q_{max}

➤The more time a vehicle spends in the atmosphere, the less accurate it will be. Thus, to increase accuracy, we use fast, steep re-entry trajectories.

➤To increase the size of the re-entry corridor, we decrease the re-entry velocity and flight-path angle.

However, this is often difficult to do.

➤Table below summarizes the trajectory trade-offs for re-entry design $\dot{q}$

Figure 20 Re-entry of VN Space Capsule into earth

➤We can meet mission requirements on the design front by changing

• Vehicle size and shape, BC

• Vehicle thermal-protection systems (TPS)

➤Increasing the vehicle's ballistic coefficient, BC,

• Doesn't change its maximum deceleration, a_{max}

• Increases its maximum heating rate,

➤There are three types of thermal-protection systems

• Heat sinks—spread out and store the heat

• Ablation—melts the vehicle's outer shell, taking heat away

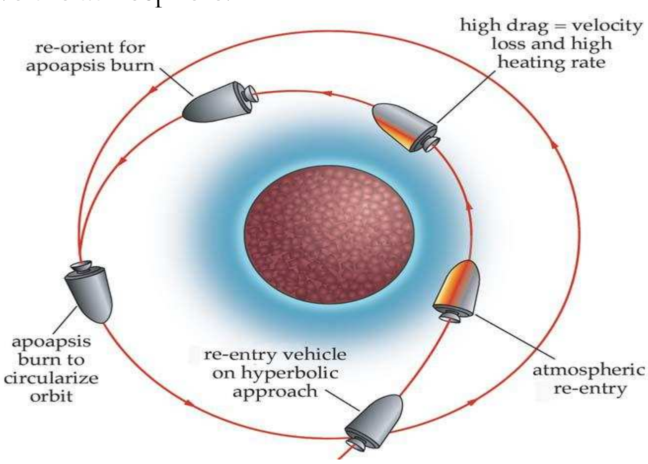
• Radiative cooling—radiates a large percentage of the heat away before the vehicle can absorb it $\dot{q}$

➤ Applying lift to the re-entry problem allows us to stretch the size of the re-entry corridor and improve accuracy by flying the vehicle to the landing site.

13.1 Aerocapture and Aerobraking

Aerobraking can greatly decrease the amount of mass needed for interplanetary transfer. During an aerobraking maneuver, the vehicle dives into the target planet's atmosphere, using drag to slow enough to be captured into orbit.

On an interplanetary transfer, the spacecraft approaches the planet on a hyperbolic trajectory (positive specific mechanical energy with respect to the planet). During aerobraking, it enters the atmosphere at a shallow angle to keep maximum deceleration and heating rate within limits. Drag then reduces its speed enough to capture it into an orbit (now it has negative specific mechanical energy with respect to the planet). To "pull out" of the atmosphere, it changes its angle of attack, lift. Basically, the vehicle dives into the atmosphere, and then "bounces" out. In the process it loses so much energy that it is captured into orbit. This atmospheric encounter now leaves the vehicle on an elliptical orbit around the planet. Because periapsis is within the atmosphere, the vehicle would re-enter if it took no other actions. Finally, it completes a single burn, much smaller than the ΔV needed without the aerobraking to put the vehicle into a circular parking orbit well above the atmosphere.



The purpose of the aerocapture before the reentry is to reduce the g-loads on the capsule and the crew. Studies looking at reentry trajectories from Mars have attempted to define acceptable effects on the crew, and in particular, the g force astronauts can withstand after long duration spaceflight.

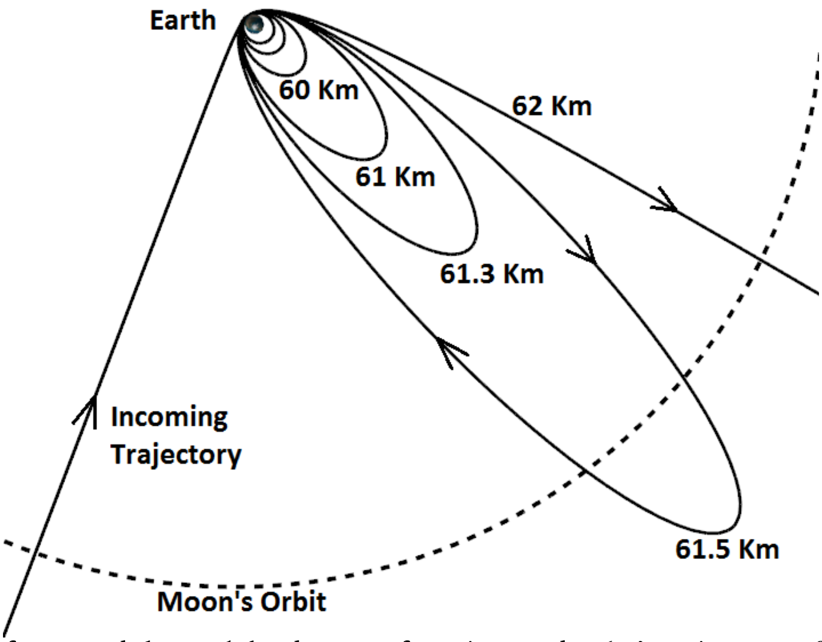


Fig 21. shows that the velocity (dashed line, with right-hand axis) during reentry peaks at about 14.2 km/sec velocity. This could possibly be reduced by changing the launch trajectory or Mars flyby conditions, which we will look at in future studies.

the g force due to atmospheric drag during aerocapture for the range of perigee altitude from 56.5 km to 62 km. The lowest perigee has the highest g force, and the peak load ranges from just under 6 g's to just over 9 g's. Of course, an aerocapture increases the length of the mission, up to an additional 10 days or so if the apogee reaches to lunar orbit. Because the service module of the capsule would be released before aerobraking, the power system of the capsule would likely rely on batteries. The post-aerobraking orbit must be optimized with these considerations, as well as with the reentry conditions. After jettisoning structures that must be released before reentry, the spacecraft is estimated to be 5,000 kg. The reentry of a 5,000 kg, 3.6 m diameter spacecraft into Earth's atmosphere present some challenges from an aerodynamics, aerothermodynamics, and thermal protection system (TPS) perspective. The mission calls for both an aerocapture maneuver and a reentry at Earth. To date, no aerocapture maneuver of this type has been attempted either at Earth or other planetary destinations. The atmospheric entry speed for the aerocapture is estimated at 14.2 km/sec, which would make it the fastest reentry of any manned vehicle by far. The fastest, successful reentry of a man-made, but unmanned, vehicle to date was the sample return capsule for the NASA Stardust mission, which reentered at 12.6 km/s with an estimated total stagnation heat flux of 1,200W/cm2.

Entry velocity, km/s	Convective heat-flux	Radiative heat-flux	Total heat-flux	Stagnation pressure
11.5	480*	30*	510*	0.33*
12.9	940†	90†	1030†	0.38†
15.0	1250†	360†	1610†	0.42†

Aerothermal environment conditions for a Stardust-sized entry capsule(-8º entry flight path angle)

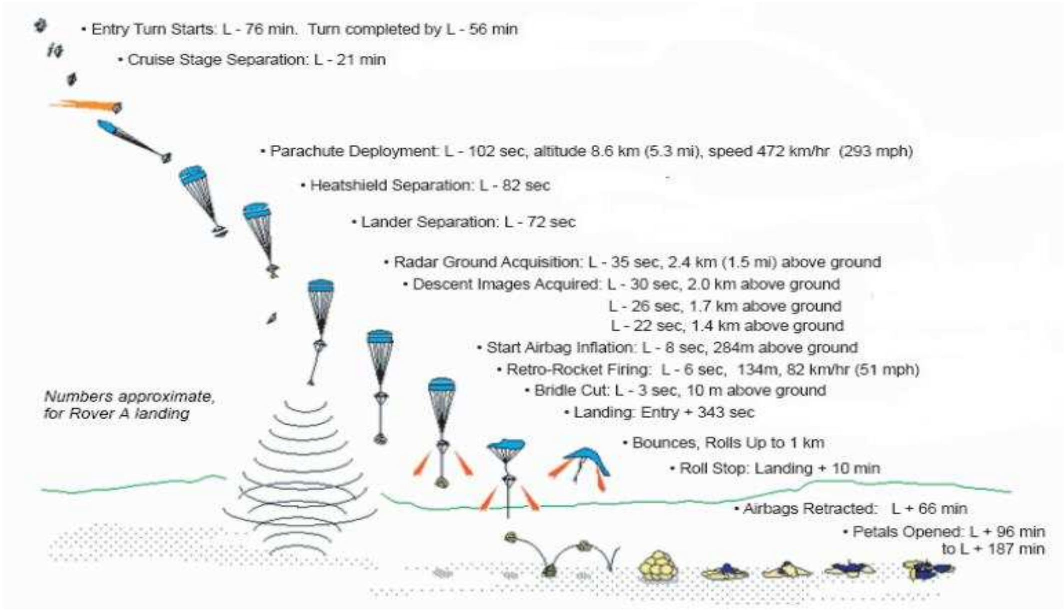


Figure 22 EDL — Entry, Descent and Landing

14. Mass estimation and cost estimation for Mars Mission 2018

The VN spacecraft would be 10 m in diameter and 25 m long. It would be spun to generate artificial gravity, with fixed solar panels generating 5 to 250 kilowatts of power. Total mass would be 25.9 metric tons, broken down as follows:

- Habitation Structure - 10.0 metric tons
- Life support system - 8.0 metric tons
- Consumables (water, food, oxygen for 1200) - 9.7 metric tons
- Electrical Power (100 kWe solar) - 2.0 metric ton
- Reaction Control system - 0.5 metric ton
- Communications and Information Management - 0.2 metric ton
- Science Equipment – 1 metric ton
- Crew - 0.2 (2) metric ton
- EVA Suits - 0.4 metric ton
- Furniture and Interior - 1 metric ton
- Re-entry Capsule - 5.0 metric tons
- Spares and Margin (25 percent) – 8.52 metric tons

Mission trajectories were calculated:

Leg	Stay Time (days)	Depart		Arrive		Flight Time (days)
1		Earth	JAN 5, 2018, 7.1756 hours GMT Julian Date 58123.7990	Mars	AUG 20, 2018, 7.8289 hours GMT Julian Date 58350.8262	227.0272
2	0.0000	Mars	AUG 20, 2018, 7.8289 hours GMT Julian Date 58350.8262	Earth	MAY 21, 2019, 20.9618 hours GMT Julian Date 58625.3734	274.5472
					Total Duration	501.5744

The total estimated cost of $ 5.821 billion broke down as follows:

- Habitation development using ISS technology: $800 million
- Re-Entry Capsule, adapted from ISS ACRV: $200 million
- Operations Costs: $250 million

- Angara Shuttle Launches: $1600 million
- Reserves and Contingency (18%) = $658 million

VN Spacecraft Mission Summary:

- Summary: Mars flyby with manned spacecraft hovering at L1 Sun-Mars Lagrangian point.
- Propulsion: ADVANCED LOX/LH2,SEP
- Braking at Mars: propulsive
- Mission Type: lagrangian
- Split or All-Up: all up
- ISRU: flyby
- Launch Year: 2018
- Crew: 2
- Outbound time-days: 227.0272
- Return Time-days: 274.5472
- Total Mission Time-days: 501.577
- Total Payload Required in Low Earth Orbit-metric tons: 100
- Total Propellant Required-metric tons: 54
- Propellant Fraction: 0.54
- Mass per crew-metric tons: 50
- Launch Vehicle Payload to LEO-metric tons: 45
- Number of Launches Required to Assemble Payload in Low Earth Orbit: 1
- Launch Vehicle: ANGARA V7

Characteristics

Unit Cost $: 820.000 million. *Crew Size:* 2.

Gross mass: 40,100 kg
Height: 25.00 m

Conculsion

In this report Vimana Notion Design Team proposed the conceptual design of a Spacecraft for Mars flyby mission 2018. The salient features of the VN spacecraft is two persons can be accommodated in the spacecraft to visit the mars and comeback to earth without landing on mars. The VN Spacecraft consists of Crew transfer module, Service module, Habitat module and Hybrid propellant module. The VN Spacecraft has main systems, subsystems, and auxiliary systems for the Mars travel. The main systems are the Power and its distribution systems, Environmental Control and Life Support System (ECLSS) for Mars mission, Anti-radiation shield systems. Each main system has subsystems to take care of long duration manned missions, good shielding and auxiliary systems to take care the complete mission requirements like characteristics of liquid hydrogen fuel and water,etc. VN Spacecraft is equipped with the habitat module, Crew Transfer module, Service module and propulsion module for accommodations and provisions for crew members of average aged couple and Scientific payloads for the spacecraft.The Anti radiation shield is established with torus-solenoid rings method which is lighter than other methods and the effect of this quadrupole magnetic field on energetic particles is stable in order to shield spacecraft during Mars mission. The Thermal Protection System (TPS) and Systems integration to the launch vehicle also discussed in detail. The Angara K7 (Russia) heavy launch vehicle can be used to launch the VN Spacecraft with the weight of 43,000kg approximately to LEO. The safety to the crew members, emergency escape system and all other systems are explained in detailed in this report. The technologies used for development of VN Spacecraft is feasible and can be ready for January 2018 mars mission.

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