Propulsion, power and thermal systems for a Mars mission

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Abstract

The aim of this paper is to prove the feasibility of a Mars mission from the propulsive and electrical generation point of view. Different technologies are presented, while calculations are performed for fuel mass and engines/subsystems main dimensions. Bimodal nuclear thermal rockets, along with solar arrays and batteries, has been proven to be the best option for both propulsion and power generation during the whole mission. Safety requirements and thermal control system are also studied.

I. Introduction

Colonizing Mars is becoming an increasingly popular topic in today's space research. Several space agencies and private companies are investing huge amounts of money in the study of advanced technologies that would enable a future landing on Mars. However, rocket engines now available for space exploration are not able to simultaneously achieve high specific impulse values and, at the same time, consume little propellant. It is therefore necessary the development of new propulsion technologies; important requirements of the propulsion system could be, for example:

- high specific impulse;
- high level of thrust;
- low risk of failure to safeguard crew on board;
- bimodal operation for primary power generation.

During the past years several solutions have been studied: the most intriguing is undoubtedly represented by the Nuclear Thermal Rocket (NTR), designed and developed by NASA as part of NERVA program[1]. This technology, in particular the bimodal NTR, is being studied and analyzed in this paper. Due to its properties, it has been found to be the most suitable for a Mars mission.

A brief description of the Mars mission, together with its requirements, is given in section II.
A detailed overview of the NTR propulsion system, along with other available technologies, is instead shown in section III. Engines and fuel mass estimations for the mission are instead discussed in section IV. Electrical power is provided by the nuclear rocket due to its bimodal operational mode: an extended overview, together with power requirement calculations for interplanetary travel and Mars/Earth orbit, is instead discussed in section V. Thermal control system is instead the topic of section VI.
Last two sections are related to safety issues (section VII) and off-nominal cases (section VIII).

II. Mission requirements

Several constraints have been considered for the manned Mars Mission outlined in this paper. Firstly, it is assumed that the space ship adopted to travel backward and forward from Mars should be able to carry 30 passengers and provide them all life support facilities.

The whole ship should be ready by the year 2032. Optimistically, it is assumed that by then already

Table I: Fuel masses comparison, Nuclear & Methane

	Nuclear thermal, tons	Methane, tons
Earth to Mars	237	1529
Mars landing	27	104
Mars ascent	19	66
Mars to Earth	304	1853

several human missions to Mars have been performed since some years. Furthermore, the ship should be built for many trips back and forth between the planets.

The mandate of the propulsion and power systems team was: to provide the means of transporting the 30 passengers and 8 crew from LEO to Mars, a mechanism for reaching Mars surface from Mars orbit and a way by which they can then return to Earth. Team agreed also to include an artificial gravity system on board in order to offer an Earth-like condition to the crew. The engine technology that best fit this mandate had to be identified and the engines sized to provide the necessary thrust for the mission. Also the power budget for various stages of the mission was calculated, the logistics for the transitions between primary and secondary power systems evaluated and thus the primary power generation equipment sized. Full redundancy of the power system was essential and it was paramount to choose the best compromise between performance and safety.

III. Propulsion Technology

The propulsive method chosen is nuclear thermal. The main points of concern while selecting the propulsive system were: efficiency, thrust, thrust-to-weight, convenience and the stage of development of the technology. The system had to be efficient enough to avoid bringing a huge amount of fuel for the mission. Having massive quantities of fuel would make for a complex and costly mission. The thrust of the system had to be high enough to be able to takeoff from Mars ground to Mars orbit. This was the critical stage of the mission: the stage requiring the most power from the engines and therefore the point for which the engines were sized. Less importantly the thrust also had to be high enough to reduce the burn times. A high system weight was detrimental to the mission, especially if it were to have to land and takeoff again. The convenience criteria were as follows: the fuel used, the ease of refueling, price, the safety and the possibility to use the same system for all phases of the mission. The stage of development of the technology refers both to the track-record and proven performance of the technology as well as whether any large scale up or development is required.

For this mission, several technologies were studied: nuclear thermal (NTR), magnetoplasma, and chemical (methane). The chemical option was seen to be very convenient due to its refueling potential, having a very high thrust to weight, as well as being a proven technology with no need for a large development. However it was eventually discarded as its has relatively low efficiency. The specific impulse of these engines can go up to roughly 350s, which is fine for an ascent vehicle, but too low for long burns in space. The fuel masses required are compared to the nuclear thermal solution in Table I. (All the values are calculated for the worst case scenario, assuming the Venus flyby trajectory, no aerobraking and a fully powered descent.)

As can be seen here, chemical rockets are not a practical option due to the huge amount of fuel required. For this reason chemical technology was discarded.

The magnetoplasma option was discarded because the thrust was too low to be used for the descent and ascent phases, so it would have to be coupled with another propulsive system for that part of the mission. This would raise the already prohibitively high mass of the system, and increase the cost further. The

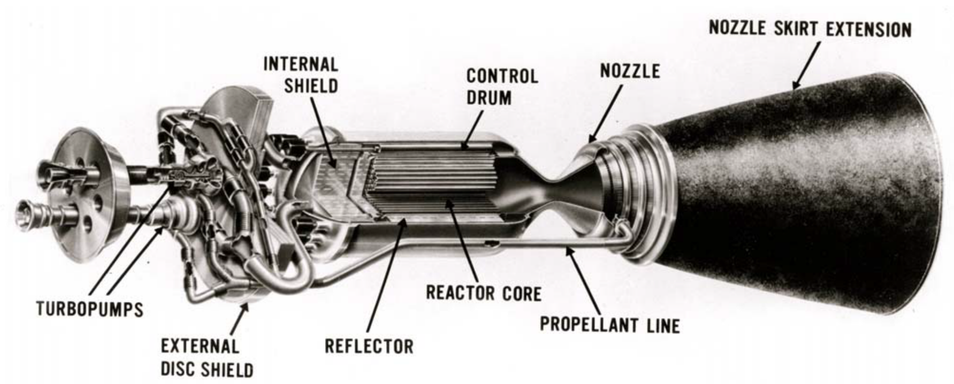

Figure 1: Schematics of an ”Expander cycle” nuclear thermal rocket engine. (credits: NASA)

technology is also very new and underdeveloped (the only actual propulsor under study is VASIMR) and has not been proven for a mission of this scale. Too much research would be required to before the imminent deadline of 2035 for this to be considered seriously, especially compared to the nuclear thermal solution, that has been the developed half a century ago. The nuclear option is much cheaper because of this, and the technology is already for missions.

The thermal nuclear option was chosen in the end because of an overall good performance in all criteria. The thrust is high enough to reach Mars orbit from ground, with a global thrust to weight of 5,5 for the engines alone. The specific impulse is very high, the (very conservative) value used in this paper is 900s. This means that the fuel masses can be kept to a relative minimum, simplifying the overall mission. The system could therefore be used for the space and ascent-descent phases of the mission. The system is also very convenient to use, at it only uses one fluid, liquid hydrogen, that can be produced on Mars via electrolysis. Being able to refuel on Mars greatly reduces the mass and price of the mission. All the is really required to produce the fuel on Mars is power which can come from the solar panels assumed to be there meaning Martian fuel is effectively free. This also means that the system is very safe, as there is no risk of explosion, since the hydrogen has nothing to react with. Coupled to the fact that nuclear reactors are very safe and reliable, this system was logically the best possibility for a long mission.

Also, the NTR technology requires absolutely not scale-up. The NERVA/Rover programs proved that these rockets are capable of all of the requirements for a Mars mission: they have been tested for the required Thrust, Exhaust temperature, burn duration performance levels as well as restart capability. Over 20 rockets have been tested at the required performance levels, so there is a strong evidence for repeatability. And this technology was developed mainly in the 60’s and then remained relatively undeveloped for a long time, there is clearly much room for progress with this, so while for this mission no scale up is required, the potential for growth of the technology and even better propulsion systems to be developed is obviously available.

IV. Propulsion Specification

The system was dimensioned by taking the amount of thrust and electricity needed and dividing it up using the power rating for existing Nuclear Thermal Rocket technology to determine the number of engines required, as opposed to scaling up or down existing technology. This determines the weight of the system, the thrust, and the fuel masses. Each engine has a thrust of 110 kN and a mass of 2,04 tons. The first requirement was the ability to ascent from Mars ground to orbit. At this point in the mission, only the shuttle is used, with an empty mass of 35 tons without engines. To obtain a thrust to weight ratio, 5 engines were used, giving a thrust of 550 kN. The added engine mass brings the empty mass of the

Table II: Thrust to weight ratio of the shuttle

	Thrust to weight ratio
Minimum (Mars descent)	$2,82$
Minimum (Mars ascent)	$2,92$
Maximum (Empty fuel)	$3,24$

Table III: Thrust to weight ratio of the shuttle

	Delta-v per maneuverer [km/s]
Earth orbit to Mars flyby	$3,58$
Mars flyby to Mars orbit	$2,4$
Mars orbit to Mars ground*	$4,1$
Mars ground to Mars orbit*	$3,11$
Mars orbit to Earth flyby	$4,15$
Earth flyby to Earth orbit.	$3,64$

shuttle to 45 tons, and the fuel needed to get to orbit weights 19 tons, as seen in Table I. Table II shows the thrust to weight ratio information of this configuration.

These thrust to weight ratios are very high, and giving a good leeway in regard to possible maneuvers, and give a very high redundancy in the case of an engine failure. The fuel masses were then computed for the whole mission and shown in Table II. To calculate the fuel masses, the delta-v map provided by the operations group was used (Table III). The worst case scenario was also assumed for these, ans the values marked with an "*" are for the lander only, as the habitat stays in orbit. The other values are for the whole ship.

V. Power generation

Electrical power is generated on board by means of five bimodal nuclear thermal rockets installed in the lander module. These engines, other than providing propulsion to the space ship itself, can be also used as electrical power generators. The lander (and its engines) are detached from the orbiting module during both the stay on Mars and the initial phase of the mission so an auxiliary generation system composed by solar arrays and batteries provides the amount of power required to run the subsystems during these critical phases. This auxiliary power generation system only has to provide 'skeleton power', i.e enough to keep power to run fundamental systems for example basic thermal control so that the components aren't damaged when the main engines return.

A. Power budget

Power budget has been calculated considering all input coming from all mission subgroups. Figure 2 illustrates the amount of power required to run every subsystem of the space ship during the interplanetary travel; hereafter this case will be referred as the nominal one.

Power estimations have been performed by each mission subgroup and are illustrated in Table IV. Power requirement are expressed in terms of kW. Values are obtained from calculations and preliminary assumptions conducted using data from other Mars missions and ISS (see paper of other subgroups).

As a result, an amount of 48.91 kW needs to be provided by the spacecraft power generation system during nominal operation. Peak power has instead been used for sizing the system: the latter has been calculated through a safety factor of 1.5 and takes into account off-nominal cases, like life support system or computer overloads and emergency cases. However, such a huge amount of power needs not to be

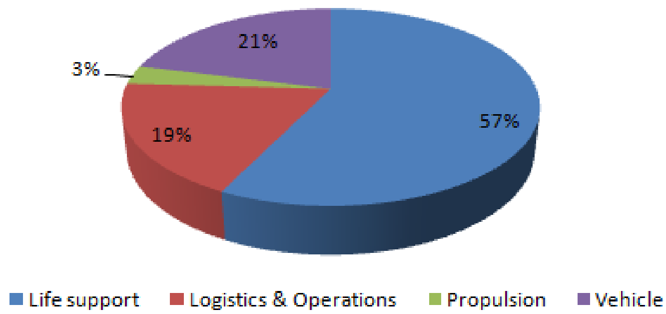

Figure 2: Percentage of power required buy each mission compartment.

Table IV: Power budget estimation for the interplanetary travel.

	Required power (kW)
Life support	$28,01$
Logistics & Operations	$9,1$
Propulsion	$1,5$
Vehicle	$10,3$
Total nominal	48.91
Total peak ($\times 1.5$)	~ 75

generated during the orbital phases around Mars and Earth. Since no crew is expected to be on-board during these phases, life support system functionality can thus be shut down, a part for some primary functionalities like humidity control and fire detection. Table V shows preliminary estimations of the power budget during off-nominal phases: estimations have mainly been performed through comparisons of different Mars orbiters[8][9].

B. Electricity generation

The nuclear thermal rocket technology which will be the primary propulsion mechanism can be constructed to be bimodal in nature (Bimodal Nuclear Thermal Rocket- BNTR). In its primary mode the fission of Uranium 235 atoms provides the energy to superheat the high pressure liquid hydrogen and which is then expanded through a supersonic nozzle to deliver thrust.

In its alternative configuration it can also be a generator of electrical power. In this design some of the energy generated in the reactor fuel assemblies is removed in a closed loop coaxial energy transport duct carrying He-Xe coolant. This heated gas is routed through a Brayton rotating unit leading to $25kW_e$.

There are several advantages of this power source. The opportunity to capitalize on mass already on

Table V: Power budget during off-nominal phases (Mars/Earth orbit).

	Required Power (kW)
Payload Instruments	0
Comms	1.2
Data Handling	0.6
AOSC	0.5
Propulsion	1
Power	1
Thermal	4.8
Life Support	1
Total	**10.1**

board the craft is extremely useful, and these engines provide very high power output with only massive solar arrays or solar dynamic cycles capable of providing the equivalent. Of these, only nuclear is useful irrespective of the distance from the sun, so while it would be possible to power the spacecraft via solar arrays, it would be placing unnecessary limitation on the ships capability, as solar power would not be effective past Mars. So, not only is nuclear the optimum choice for this mission but it is also the logical technology to develop when missions beyond Mars are attempted. Also due to the source being nuclear, the ship does not have to account for eclipses and carry cumbersome batteries for secondary power for the whole ship.

The BNTR technology in this current architecture is only the 1st generation technology. 2nd generation BNTRs are currently being developed that are expected to be capable of producing between 100kW_e-1MW_e. This is monumentally powerful and the electricity produced by each engine outstrips demand for power for the whole spacecraft by over an order of magnitude.

C. Power Generation during Orbital Phases (PGOP)

Auxiliary power is required during the orbital phases for the skeleton power: solar panels and batteries are sufficient to fulfill this requirement. Unlike for the main power system with skeleton power they do not need to provide a massive power output, this was their limiting factor as a primary power source.

1) Daylight and eclipse:

While orbiting around Mars or Earth, it is important to know whether the solar panels are hit by the solar rays or shaded due to planet umbra. During eclipse phases the solar arrays cannot be operative and power generation functionalities need to be delegated to batteries.

Moreover, solar constant plays a key role for sizing the auxiliary power system. Solar constant value on Mars is $591\ \mathrm{W/m^2}$, less than half of the corresponding value on Earth ($1367\ \mathrm{W/m^2}$). This clearly states that the scenario to be considered when sizing the PGOP system is the orbital phase around Mars as that will be the point at which the power system will be least efficient.

Orbit altitude around Mars has been estimated in $400\ \mathrm{km}$; knowing Mars radius ($R_M = 3390\ \mathrm{km}$) one can obtain, from simple basic physics calculations [6], the orbital period:

$$T_M = 7076\ \mathrm{s} = 118\ \mathrm{min} \tag{1}$$

Simple geometrical relations allow to estimate the times of daylight and eclipse, respectively $T_d = 76.27\ \mathrm{min}$ and $T_e = 41.51\ \mathrm{min}$.

D. Solar arrays

A trade-off between available technologies has been performed prior to start with calculations. Multi-junction GaAs solar cell, in particular, has been found to be a proved and reliable technology. Compared to single junction GaAs, silicon or indium, multi-junction GaAs cells achieve the best compromise between high efficiency (up to 28%) and good cost effectiveness[7].

E. Batteries

Secondary rechargeable batteries are used for secondary power generation during eclipse phases.
Many different types of batteries aimed at space purposes are available. For batteries there is an obvious trade-off between cost per launch and performance for the different available technologies.

Table VI shows characteristics and performance of the most common battery technologies for space

Table VI: Batteries specification[4].

	Li-polymer	**Li-ion**	**NI-Cd**
Specific energy (Wh/kg)	$100-200$	$90-150$	$40-50$
Specific power (W/kg)	>200	$200-220$	$150-200$
Power density (W/l)	>400	$400-500$	$300-500$
Operating range (°C)	50 to 70	10 to 45	−20 to 50
Heat capacity (Wh/kgK)	$0,4$	$0,38$	$0,3$
Relative cost ({£}/kWh)	>3000	3000	1500

purposes. It clearly shows that, although cheaper, Ni-Cd batteries can not provide the same amount of energy compared to more recent and advanced technologies like Li-Ion and Li-polymer. Although very promising, this latter technology is still proven not to be a reliable technology in space[10], mainly due to its high-temperature operating range. As a result, we selected Li-Ion batteries which compared to Ni-Cd batteries presents several advantages:

- low weight due to the use of lithium;
- higher specific energy;
- lower auto-discharge rate;

F. POPG dimensions

The area of solar arrays have been estimated by keeping under consideration several parameters like efficiency, eclipse duration, efficiency of electrical paths from source to loads, inherent degradation[1] and life degradation[2].

A reasonable estimation of the power provided by the whole system of solar arrays, to power the spacecraft for the entire orbit, can be found through the following formula:

$$P_{sa} = \frac{1}{T_d}\left(\frac{P_e T_e}{X_e} + \frac{P_d T_d}{X_d}\right) \tag{2}$$

where P_d and P_e represents respectively the necessary power to be generated during daylight and eclipse and X_d and X_e the corresponding factor which takes into account path losses between source and loads: these latter two values have been set to 0.6 and 0.8[10]. P_e and P_d has been calculated through a proportion between the value that comes out from the power budget when the space module is orbiting around Mars and the two time durations T_d and T_e: the calculation leads to a total required power of $11.40\ \mathrm{kW}$.

Degradation factors for a 27% GaAs solar panel are here reported:

- inherent degradation: 0.77
- degradation due to radiation: $2.75\%\ \mathrm{degr/year}$
- life degradation factor for a 932 days mission: 0.9313

Knowing these parameters allows to calculate the array performance per unit area at end of life (EOL), estimated as $118.70\ \mathrm{W/m^2}$. Ratio between EOL power per unit area and required power can be used to calculate the area of the solar array: for our system, this area has been estimated as $98.06\ \mathrm{m^2}$. This also means that for a system with a specific energy of $200\ \mathrm{W/kg}$ one can get a mass for the solar power generation system of 5.48 tons.

[1]Inherent degradation includes degradation of solar cell efficiency due to assembly of the solar panel components, temperature of operation of the solar array and shadowing of cells.

[2]Life degradation occurs because of thermal cycling in and out of eclipses, radiation environment, micro-meteoroid strikes, plume impingements from thrusters, and material out-gassing

Batteries have been sized instead considering transmission efficiency between source of power and loads, eclipse time during an orbit and specific energy. The following formula can be used for calculating the mass of the battery pack:

$$M = \frac{P_e T_e}{E_{sp} DOD \eta} 1.5 \tag{3}$$

where the safety factor 1.5 has been added in order to take into account the eventuality of failure of one or more battery modules. Considering that batteries maintain better their performance when discharged only a small percentage from complete charge, depth of discharge (DOD) is set to 25%. Knowing all parameters, the total mass of the battery pack has been estimated as 117.27 kg.

VI. Thermal control system

The purpose of the spacecraft thermal control system is keep all the modules of a spacecraft working within acceptable temperature ranges during the mission. Based on the theory of thermodynamics and heat transfer, thermal control can be achieved by controlling the heat exchange processes between inside and outside of a spacecraft. In other words, the following equation should be satisfied:

$$Q_{in} - Q_{out} = C \frac{dT}{dt} \tag{4}$$

Q_{in}, the heat absorbed by the spacecraft, including radiation from sun and other planets, albedo, thermal power generated by engine and heat produced by devices and human metabolic. And Q_{out} is the heat should be rejected into the space in order to keep thermal equilibrium. The heat exchange between spacecraft and space environment is dominated by radiation. The characteristics to describe the thermal control system are:

- Environment interaction
- Heat collection
- Heat transport
- Heat rejection

The following Figure 3 shows heat flow of a thermal control system. Generally speaking, thermal control is a result of synchronous work of passive thermal control system (PTCS) and active thermal control system (ATCS).

1) Passive Thermal control System(PCS):

Passive thermal control itself has no ability to automatically adjust the temperature, but relies on the selection of materials and a reasonable arrangement for heaters, heat pipes and cooling loops, it is able to deal with the heat exchange, not only between inside and outside, but also between different modules. For example, coatings of external surfaces and internal multi-layer insulation (MLI) are commonly applied.

Coatings are the simplest and least expensive of the TCS techniques. A coating may be paint or a more sophisticated chemical applied to the external surfaces of the spacecraft to lower or increase heat transfer. The characteristic of the type of coating depends on their absorptivity, emissivity, transparency, and reflectivity. The main disadvantage of coating is that it degrades quickly due to the operating environment[11]. Coating materials well applied nowadays are silicon oxide (SiO_2) and Aluminum oxide (Al_2O_3), those materials can increase reflectivity and decrease absorptivity.

Multilayer insulation (MLI) is the most common passive thermal control element used on spacecraft. MLI prevent both heat losses to the environment and excessive heating from the environment. Spacecraft components such as propellant tanks, propellant lines, batteries, and solid rocket motors are also covered in

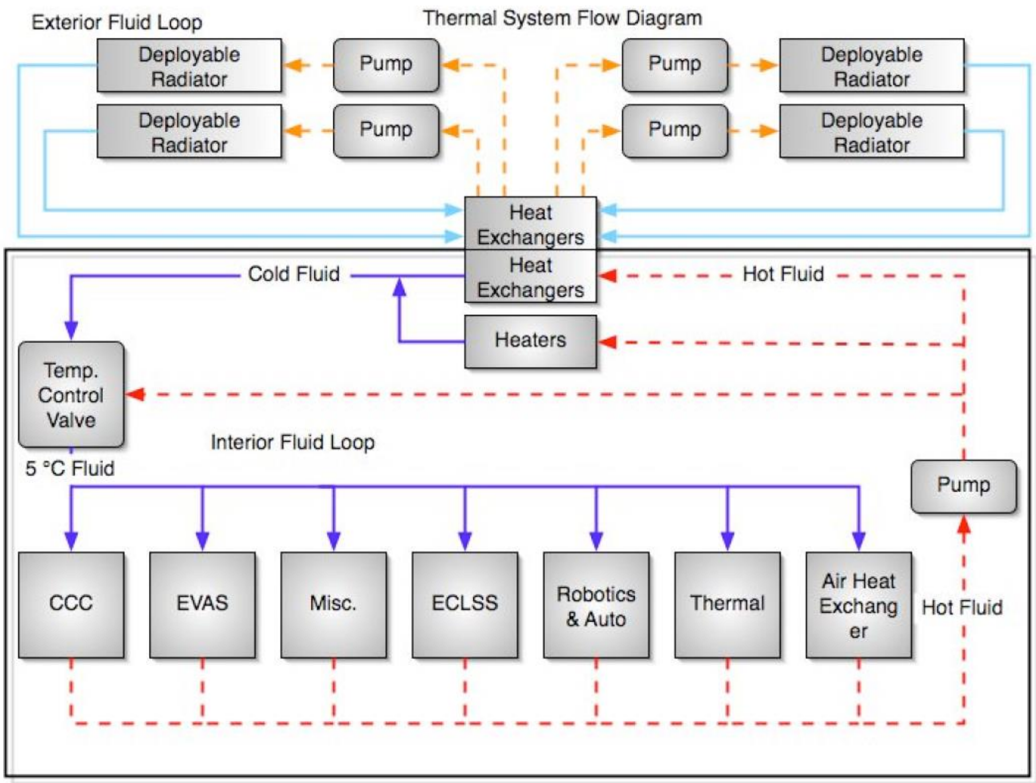

Figure 3: Thermal system flow diagram

MLI blankets to maintain ideal operating temperature. MLI consists of an outer cover layer, low emittance interior layer, and an inner cover layer. The outer cover layer needs to be opaque to sunlight, generate a small amount of particulate contaminates, and be able to survive in the environment and temperature to which the spacecraft will be exposed to[11].

Heat pipes and cooling loops are working in similar way and both of them are based on conduction and convection. A reasonable arrangement for heat pipes and cooling loops can transport relatively large quantities of heat from one location to another without electrical power.

2) Active Thermal control System(ACS):

The operation temperature range varies for different devices in the spacecraft. Furthermore, for a Mars mission, the temperature in the cabin should be precisely controlled within an acceptable range for everyone. Thus, sometimes only passive thermal control measures are not sufficient enough due to some design limitations. The solution is to apply an active control system working synchronously.

Louver is a device with low emissivity blades which can reject heat emission from high emissivity instruments. The most commonly used louver is the bimetallic, spring-actuated, rectangular blade louver also known as venetian-blind louver. Louver radiator assemblies consist of five main elements: baseplate, blades, actuators, sensing elements, and structural elements[11].

The most important equipment of an active thermal control system for temperature equilibrium control is the radiator system. Due to space environment, the only way to reject excess heat into space is radiation. The system consists of many deployable panels which means the total area of deployed radiators depends on the working situation. Inside each panel, there is a series of tubes that are routed throughout the radiators. For the Mars mission, Hydrogen flow contain large quantity of excess heat can be controlled by pump and flow through the radiator tubes. Flows inside those tubes are generally can be classified in two kinds[11]:

- single-phase loops, controlled by a pump
- two-phase loops, composed of heat pipes (HP), loop heat pipes (LHP) or capillary pumped loops (CPL)

With interaction between radiators and the space environment, excess heat could be rejected. The radiator size can be expressed as the following equation:

$$A = \frac{Q_{out}}{\sigma \epsilon \eta (T_r^4 - T_e^4)} \tag{5}$$

Where Q_{out} is heat rejected per unit of time, σ is Stefan-Boltzmann constant, ϵ is emissivity of the radiator which is defined as 0.9 in this case, η is radiator efficiency equal to 0.85, T_r and T_e are radiator operating temperature and environment temperature.

According to calculation, the radiator size is roughly $1100\ m^2$, in this case, two-sided deployable radiator panels are chosen to allow the radiators to be autonomous, self-deploy/activate upon arrival, and to reduce the mass and actual area of the panels. The radiators can have similar design as those being used in ISS, each radiator consists of seven panels (each about 6 by 12 feet)[12].The majority of radiators for the Mars mission are attached to engine module and some of them are connected to device module.

VII. Safety

There is some stigma that come with nuclear power that it is more dangerous than conventional sources. In the case of rocketry it is actually the reverse, with NTR technology possibly being the safest available technology in terms of potential for catastrophic damage. This is due to the lack of an oxidizer which means that there is not hundreds of tons of combustible substance on board, significantly reducing the potential for explosion. The engines will not be utilized to help reach LEO and will instead be sent into LEO and assembled 'cold' with the fuel rods not inserted in the reactor. So, any catastrophic problems that would cause the rods and reactor to crash onto earth or in orbit would have minimal impact. The nuclear reactor is only potentially very dangerous when 'hot'. The BNTRs are only used in three scenarios; the first is as the workhorse for interplanetary journeying which is where there is the lowest probability of engine failure.

The other potential opportunities for failure of the engines is when the lander separates from the hab and lands on Mars, or during the ascent of the lander back to the hab. However the vibrational and acceleration forces that the engines will undergo are much lower than the equivalent journey on earth.

If these are the stage of the journey where catastrophic accidents are most likely to occur, it is sensible to assess the potential impact of a worse case scenario during a launch/landing on Mars. In 1965 NASA conducted the KIWI TNT test[1] where they allowed one of these reactors to go supercritical to assess the potential for damage in a worst case scenario. The result was that the radiation fallout would cause fatalities up to 600 feet and injuries up to 2000 feet. The true potential for damage comes from if the reactor goes supercritical in an unstable orbit, at high altitude and then crash lands at extremely high velocity which would cause the radiation to have a wider impact. This would be an unacceptable risk on earth but on Mars it is more acceptable, due to the probability for the reactor to go supercritical early enough for the velocity to be high and to crash land near the Martian colony is extremely unlikely. It would be sensible to place the launch and landing site a minimum of 2000 feet from a Martian colony, although more research into simulation of supercritical reactor crash landings could provide a statistical choice for an adequate minimum distance between landing/launch site and the colony.

As the mars lander does not carry the responsibility of shuttling fuel to the main habitat in orbit it avoids the greater likelihood of failure associated with greater amounts of cyclical loading. This will be the responsibility of a staged chemical rocket already on Mars, which will have the sole job is refueling.

VIII. Off-nominal case

1) Propulsion:

The strongest limiting factor for the propulsion module of the ship is the total thrust. The ship should always have enough thrust to ascend from Mars. This is the critical point in the mission profile, where the maximum thrust is required and hence this was the stage in the mission that the engines must be sized for. The thrust required then determines the number of engines. With the 5 engines operational, the thrust to weight ratio is about 2,82. With one engine malfunction, the thrust to weight ratio is reduced to 2,26. This is still high enough to perform the ascent phase, and is actually higher than a lot of modern rockets. For example, the space shuttle had a thrust to weight of 1,5 on takeoff, and some rockets go as low as 1,3. In the unlikely event of the ship losing another engine, the thrust to weight ratio falls to 1,69. This is still high enough to get to orbit, or land back safely. The ship is therefore resilient to two engines failing.

2) Electricity:

The redundancy for the electricity generation for the ship comes in the form of there being significantly higher potential for generation than is being utilized by the ship. As each engine produces 25kW_e and the maximum total power consumption for the ship is $48.91\ \mathrm{kW}$ (nominal) and $75\ \mathrm{kW}$ there is the possibility to lose 2 engines and still generate enough power to run the ship. The probability of 2 engines failing was considered low enough for this to be a satisfactory solution. The alternative choice of using another technology as the back-up power was deemed unreasonable as having sufficient solar arrays or batteries to provide power for the whole ship would be too much mass to warrant the increased safety. Although there is even potential for losing 3 engines and still being able to cleverly orchestrate the remaining 2 engines, solar arrays and batteries to complete the mission: in this extreme case, artificial gravity is shut down in order to allow deployment of solar arrays and sun tracking.

IX. Conclusion

Nuclear thermal rocket has been proven to be a reliable propulsive technology for a Mars-Earth manned mission. Electrical power, along with the needed thrust for orbital maneuvers, is fulfilled by using 5 bimodal engines able to provide the 75 kW of peak power necessary to run all the subsystems. 5 engines are enough to provide enough power to run the subsystems also during off-nominal operation. $98.06\ \mathrm{m}^2$ of solar panels and a battery pack of $117.27\ \mathrm{kg}$ provide power when the space module is orbiting around Mars/Earth and can also provide additional power in case of failure of 3 engines. $1100\ \mathrm{m}^2$ of deployable radiators are used to radiate out to space the heat generated by the set of 5 engines. Although public opinion is mainly not favorable to nuclear power generation, studies proved that nuclear thermal rockets are very reliable even when utilized for long-term manned interplanetary missions.

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Oliver Boom Worked on the electrical power generation system, power budget and safety issues.
Francesco Tosto Worked on the auxiliary power generation system during orbital phases.
Guillaume Tousignant Worked on the propulsion system and performed calculations on thrust/weight ratio and fuel mass.
Tong Wang Worked on the thermal control system.