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MOBILE NUCLEAR FISSION ENGINES FOR MARS SURFACE VEHICLES AND/OR AIRCRAFT

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ABSTRACT

When humans go to Mars, we should take vehicles such as "rovers" or helicopters, powered by compact nuclear fission reactor engines. This would provide portable "muscle power" for digging, e.g., in search of life or water, and for long distance mobility to explore the entire planet without having to periodically return to a fixed ground base. Reactor power would provide continuous life support, including heat to survive the frigid Martian nights.

Technical requirements for Martian mobile nuclear fission engines are discussed. The fundamental design issue is minimizing mass while providing adequate power conversion and radiation shielding.

Nuclear fission reactors have supplied more than 20% of earth's electricity over the last several years, using either natural uranium fuel or uranium slightly enriched in the fissionable isotope U^{235}. However, a compact reactor must use either Highly Enriched Uranium (HEU) or plutonium fuel in its core. A plutonium core would be smaller than a HEU core but would suffer from a smaller experience base. Compact nuclear fission reactors using HEU are ubiquitous in the submarines of several nations' navies and are even used to power some aircraft carriers.

Substantial design, development and testing activity was pursued in the 1950s and 1960s to adapt mobile nuclear fission reactors to propel jet aircraft and nuclear thermal rockets. The aircraft and rocket applications are not in use today, in part due to radiological safety issues that could accompany a crash in a populated region. In contrast to plutonium, HEU fuel carries the safety advantage of having negligible radioactivity before it is inserted into a reactor and fissioned. If a Mars mission with an unoperated HEU-fueled reactor as cargo were to crash while being launched from earth, radiological safety issues would be insignificant.

OPEN BRAYTON CYCLE

The "open Brayton cycle" provides an effective implementation of mobile engines. Atmospheric air is compressed to an elevated pressure, heat is added at constant pressure, and then the compressed, heated air expands through a turbine. Expander work exceeds compressor work and provides net power as shaft rotation torque.(1,2)

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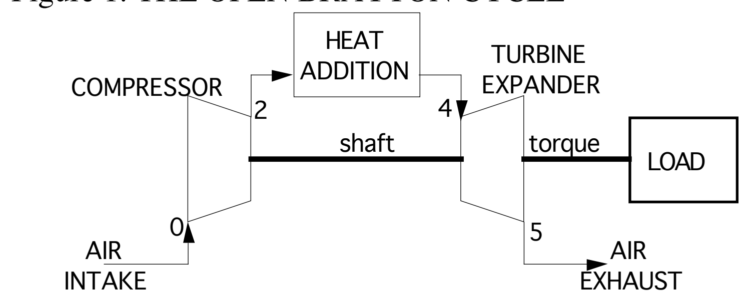
Fixed installations employing gas turbines for electric power production usually add various heat exchangers between airflows to more efficiently use heat, via regeneration and/or intercooling. But these additions are not appropriate for a mobile engine since they add significant weight. For a mobile engine, the simple Figure 1 diagram is proper.

On earth, the open Brayton cycle has been used in aircraft and surface vehicles. For such terrestrial implementations, the "heat addition" stage is usually implemented by burning hydrocarbon fuels in the compressed air. Although further temperature increase would in principle boost output power and efficiency, air temperatures are not usually designed to exceed about 1300°C due to turbine blade materials limits.

Figure 1: THE OPEN BRAYTON CYCLE



WHY WE NEED LONG RANGE MOBILITY AND NUCLEAR ENGINES

NASA's present stated cost of launching into low earth orbit is $10,000 per pound mass, i.e. $22,000/kg. Using a hydrogen-oxygen rocket to inject into a transfer orbit to Mars and making a minor allowance for aerobraking equipment and for deadweight such as fuel tanks and rocket engines, NASA's present delivery cost to the surface of Mars must exceed $71 million/metric ton. The delivery charge alone for a 53.8 metric ton Mars Direct mission would thus exceed $3.8 billion. Median US household income in 1995 is stated by the census bureau as being $34076. Thus the delivery charge for each Mars Direct mission is equivalent to the total annual income of over one hundred thousand median income US households.

Because the crew of each manned mission to Mars will spend about 18 months on the planet Mars, and because each mission will be expensive, it would be wasteful for manned exploration to be limited to the immediate vicinity of each landing site. For instance, if round-trip long distance mobility on Mars' surface were severely limited to not exceed 100 km (62 miles) from each landing site, then 4554 separate missions to Mars would be required to visit the entire planet once. A happier situation would provide round-trip long distance mobility to explore any location within a 2171 km (1349 mile) distance from each landing site, allowing access to 10% of Mars' surface area during each mission. The ultimate capability would support round-trip excursions up to 10600 km away, providing mobility to explore any place on Mars from any single landing site.

The alternative to using a mobile nuclear reactor engine would be to use a conventional engine powered by the chemical reaction of fuel and oxidizer, such as an internal combustion engine or a combustion gas turbine. But Mars has no global network of fuel and oxidizer manufacturing plants, so chemically fueled vehicle engines would need to be supported by the infrastructure of a fuel and oxidizer manufacturing plant built at a ground base. The Martian air, composed primarily of CO_2 gas, cannot support combustion, so in addition to the weight of a fuel tank, a vehicle powered by internal combustion would also need to carry another tank containing an even greater weight of oxidizer. Since Mars lacks petrochemicals to serve as chemical feedstocks, the fuel and oxidizer would need to be manufactured from available simple materials, e.g., CO_2 and perhaps H_2O. Engine-relevant quantities of oxygen practically require a cryogenic tank along with the weight of its thermal insulation and the complexity of its boil-off pressure controls. Depending on which fuel is manufactured, e.g., CH_4 (methane), the fuel itself could also practically require cryogenic storage. Long duration trips would then require further increasing the amount of cryogenic oxidizer and fuel in order to accommodate boil-off during the excursion. The maximum weight of fuel plus oxidizer plus tanks, which the vehicle is capable of carrying, would determine the vehicle's maximum range. The maximum distance of points accessible via round-trips from the ground base would be about one third of that maximum range if a safety margin is included, but could never exceed one half of that maximum range.

On Earth, cars typically can carry enough gasoline to operate for up to 8 hours at cruising speed. The oxygen consumed has a mass 3.5 times the mass of gasoline burned, so if it were necessary to also carry oxidizer within the same total weight, the time between refueling would need to be reduced below 2 hours and the driving range similarly reduced. Some long distance passenger aircraft can cruise for up to 12 hours, but that duration would be reduced below 3 hours if it were necessary to carry oxidizer along with the jet fuel. Martian excursion duration could be extended somewhat by increasing fuel and oxidizer tank sizes at the expense of payload, or by reducing engine size and power (thus jeopardizing the crew).

To safely venture even 100 km round-trip from the landing site, the one-way range should be 300 km. For a surface vehicle with an average ground speed of 20 km/hr through rough terrain, the 300 km range would require driving for 15 hours. This is probably near the limit achievable for engines powered by combustion of chemical fuel and oxidizer. Truly long-range mobility excursions lasting for days or weeks using chemical fuels and with an adequate power level appear impossible. Nuclear power appears to be necessary for long range mobility on Mars.

GENERAL REQUIREMENTS

Long range mobility implies that between returns to the ground base, crewmembers may be inside a mobile vehicle for extended excursion periods of time, measured in days or even weeks. Such excursion durations are too long for continuously wearing space suits as done in rover excursions during Apollo missions to the moon. Each Mars vehicle with long distance mobility must therefore have a pressurized crew compartment with air rebreathing life support equipment (i.e., CO_2 removal and O_2 addition), sanitary facilities, food preparation facilities, communications equipment, sleeping bunks, and storage for consumables such as oxygen, drinking water, and food. The vehicle's engine must be capable of powering the internal equipment in addition to providing mobility. Because the nighttime Martian air temperature dips to –100°C (-148°F) at the equator and gets even colder elsewhere, the vehicle's engine must also provide adjustable heat as needed to keep the crew compartment warm.

Nuclear fission has the advantage of very high energy content, about one million times the energy content of the same mass of chemical fuels. The fission of one gram of fuel provides an amount of heat equal to a 1 megawatt power level for a 24 hour day, so a one megawatt high temperature heat source can be provided continuously throughout an 18 month Mars surface stay duration by the nuclear fission of only 0.55 kg (1.2 pounds mass) of fuel. This level of fission "burnup" can easily be provided by a single compact fission core without requiring any refueling operation during the 18 month mission. If the one megawatt high temperature heat source operated a thermal conversion engine with 25% thermal conversion efficiency, the engine output would be 350 horsepower of mechanical work plus 750 kW of lower temperature "warm" heat.

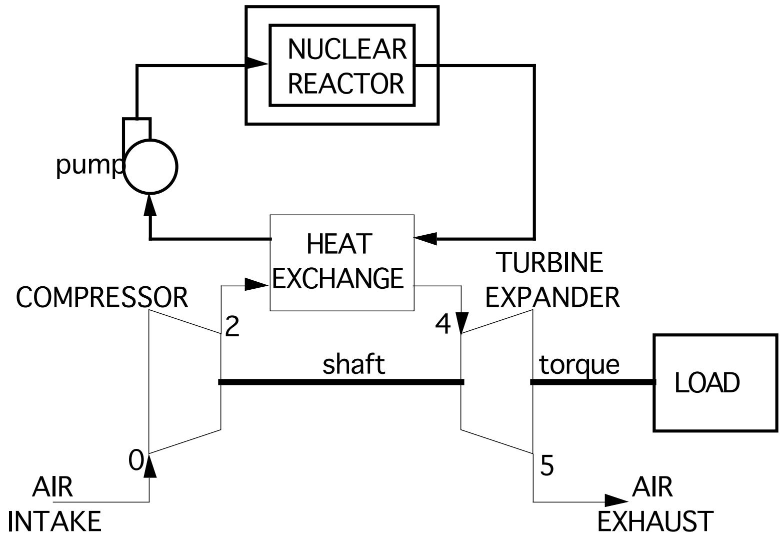
It is expected that mobile nuclear reactor engines for Mars will need to develop maximum mechanical power levels ranging from around one hundred horsepower for small surface vehicles to several thousand horsepower for aircraft. Fission (and thermal) power levels to consider for mobile nuclear fission engine design purposes therefore range from about 0.3 MW to 10 MW, depending on application details.

To save weight, the energy conversion system should be open cycle, using the Martian air as its working fluid to minimize the requirement for heavy heat exchangers. The open Brayton cycle appears to be the best choice of thermal conversion system. That, in turn, requires that the reactor have as high an operating temperature as feasible in order to avoid low conversion efficiency.

THE NUCLEAR HEATED OPEN BRAYTON CYCLE

Although possible, it is not advisable to directly heat compressed Martian air by routing it through a nuclear reactor. That would require designing the nuclear reactor with sufficient internal cooling spaces to accommodate a gas, which would increase the reactor's critical mass and prevent the reactor from having a compact size. Also, oxygen in the air's carbon-dioxide, and free nitrogen, would become activated during passage through the reactor, becoming N^{16} , a gamma ray emitter with a 7.1 second half-life, and N^{17} , a neutron emitter with a 4.2 second half-life.[8] A continuously operating engine routing air through its nuclear reactor would thus trail a short, radioactive exhaust plume.

Figure 2: NUCLEAR HEATED OPEN BRAYTON CYCLE



A good fluid to use for transferring heat between a nuclear reactor and air is the liquid metal, lithium. It has not yet been used much as a heat transfer fluid in nuclear reactors because one of its two naturally occurring isotopes, Li^6, (7.5 % abundance) has a very high cross section for absorbing neutrons and tends to prevent the fission chain reaction from proceeding. The other isotope, Li^7, (92.5 % abundance) has a very low cross section for absorbing neutrons and is compatible with use in a reactor. Happily, the two isotopes can be separated (at some cost) so that nearly pure Li^7 can be used. [4,5]

Sodium is another liquid metal that has been used extensively in certain nuclear reactor designs (i.e., breeders). However, sodium becomes temporarily radioactive within a reactor, so the heat exchanger between sodium and air would need its own gamma ray shielding for personnel protection. In contrast, Li^7 does not become activated in the reactor. The liquid/air heat exchanger for a design using Li^7 would not require shielding.

Lithium, with an atomic number of three, is a member of the chemical group known as the alkali metals. It is used in various conventional applications, e.g., in high energy density batteries, in fireworks where it imparts a crimson color, etc. Lithium melts at 180.54°C, so a lithium heat-transfer loop must be kept warmer to prevent freeze-up. As a liquid, lithium's density is about one-half the density of water. Lithium's specific heat is close to 1.0 cal/g-°C throughout its liquid range, matching the specific heat of water. But whereas water can be kept liquid up to 300°C only by increasing its pressure above 90 atm, the boiling point temperature of lithium at one atm pressure (101.3 kPa) is 1347°C.

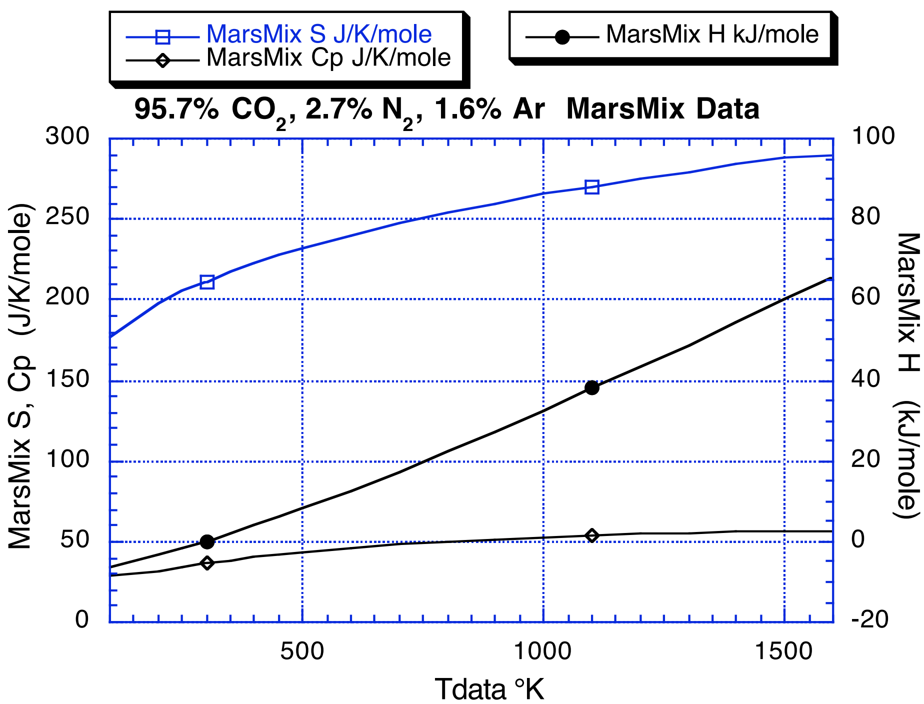
Although lithium is corrosive to many materials, tests have shown lithium exhibits long-term chemical compatibility with tungsten (a.k.a. wolfram) at temperatures up to 1300°C. Tungsten has the highest melting point (3422°C) of all known materials. It also is a very hard metal. It is used commercially in light bulb filaments and welding equipment. With present technology, it will be difficult to manufacture complicated shapes out of tungsten.

However, there are benefits to using tungsten beyond the fact that it facilitates lithium's use. Tungsten's high density (19300 kg/m^3) makes it a more effective gamma ray shield material than lead (density 11350 kg/m^3), allowing reduced gamma ray shield mass. Tungsten is more effective than lead in scattering high-energy neutrons into lower energy ranges where the moderator becomes effective. Tungsten's thermal conductivity follows closely behind the thermal conductivities of gold, silver, copper, and aluminum, exceeding thermal conductivities of other metals. Tungsten's chemical inactivity renders it a good container for fission products. And its high melting temperature might help if a reactor malfunction occurred.

PATTERN PROCESS FOR NUCLEAR HEATED OPEN BRAYTON CYCLE

The NIST-JANAF thermochemical tables[3] were used to prepare thermodynamic properties simulating Martian air, assuming its composition is 95.7% CO_2, 2.7 % N_2, and 1.6 % Ar, approximating Viking data. The calculated enthalpy (H), entropy (S), and specific heat (Cp) of this "MarsMix" versus Temperature are plotted in Figure 3. Note that a MarsMix mole's mass is 43.5 grams.

Figure 3: Calculated "MarsMix" Thermodynamic Data



Using thermodynamic relationships for perfect but nonideal gases, adiabatic process pressure changes are calculated via the relation

$$R\ln\left(\frac{P_2}{P_1}\right) = S(T_2) - S(T_1) \text{ where } R = 8.317 \quad \text{J/K/mole}$$

Given the choice of lithium and tungsten materials, the high temperature compressed Martian air emerging from the heat exchanger is assumed to be 997°C (1270°K=T_4). (This choice may be conservatively low.) The planetary average Martian air temperature is about -63°C (210°K=T_0), and is chosen as inlet air temperature, although it could vary between –133°C (140°K) and +7°C (280°K) depending on local conditions. For purposes of these calculations, the Martian air pressure was taken as 680 Pa, which is consistent with Viking data. It can be shown[(1)] that the optimum choice of open Brayton cycle compression ratio to maximize output power per unit airflow is the one which sets T_2=T_5. Using this approach, the optimum compression ratio was determined numerically from the MarsMix thermodynamic data, and found to be 112; the corresponding value for T_2=T_5 is 580°K (which also is well above lithium's freezing point temperature).

Heat flows and compressor/expander powers are obtained from the enthalpy plot. For the ideal case, heating from state 2 to state 4 adds 35.5 kJ/mole, the net work output is 20.9 kJ/mole, and the thermal efficiency is (20.9/35.5)=59 %. Actual components may have isentropic efficiencies near 80 %, as depicted by the dashed lines of Figures 5 and 6. Heating from 2' to 4 adds 31.2 kJ/mole, but the net work is 28 -18.8=9.2 kJ/mole. Calculated thermal efficiency is then a more realistic 29.5 %.

Assuming 31.2 kJ/mole heat addition from dashed line state 2' to 4, in Figures 5 and 6, a 1 MW nuclear reactor heat source would correspond to a Martian airflow of

(10^6 Watts)/(31.2 kJ/mole)*(43.5 grams/mole)=(32 moles/second)*(43.5 grams/mole)
= 1.4 kg/sec airflow

The output power produced by the engine at 29.5% efficiency would be 295 kW, equivalent to 395 horsepower.

Gas flows are not excessive inside the engines. But due to the low density of the Martian air, special designs may be required for air intake ducts. If the Martian air density is 0.020 kg/m^3, the volumetric flows at low pressure are easily calculated.

Table 1: Range of Mobile Engine Parameters

Reactor Power	AirMass Flowrate	AirIntake Volumetric Flowrate	Output Power (metric units)	Output Power (horsepower)
0.3 MW	0.42 kg/sec	21 m^3/sec	88.5 kW	119
1.0 MW	1.4 kg/sec	70 m^3/sec	295 kW	395
10.0 MW	14.0 kg/sec	700 m^3/sec	2.95 MW	3950

Figure 4:

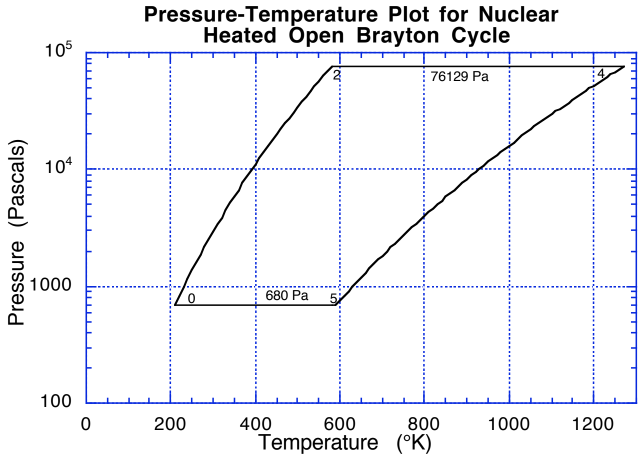


Figure 5:

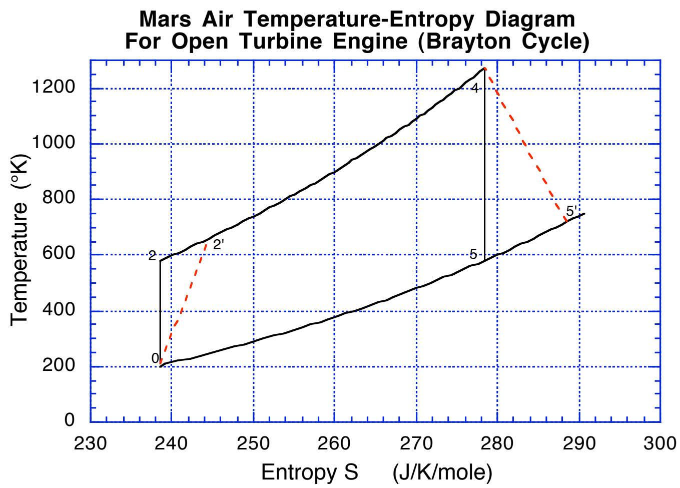
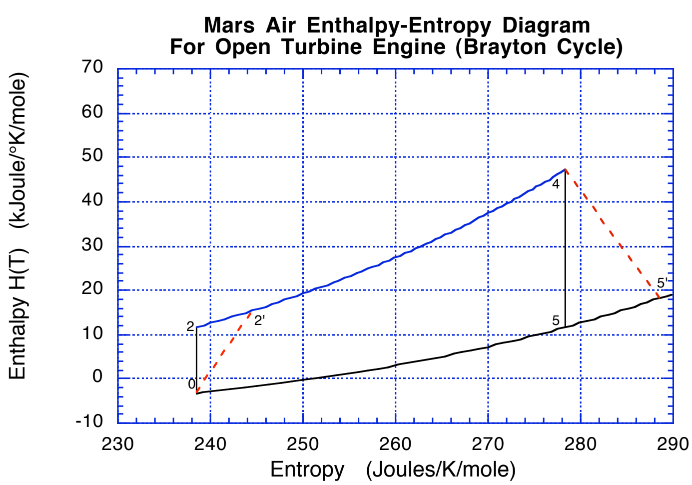


Figure 6:



LIQUID FLOW

Since this process pattern includes heating the compressed Martian air by 690°C from 580°K to 1270°K, it would be reasonable to also select a temperature difference of 690°C for the liquid lithium loop. Using lithium's specific heat and density shows that a 690° temperature change corresponds to changing lithium's heat content by $1.44 * 10^9$ J/m^3 . Required volumetric flow rates of the lithium are obtained by dividing the reactor power by this value, i.e.

Volumetric Flowrate of Lithium = (Power in watts)/($1.44 * 10^9$ J/m^3)

The total flow lithium duct cross section area is given by the volumetric flowrate divided by the designed flow velocity. A commonly used flow velocity for many cooling applications is 3 m/s. Arbitrarily selecting it, we have

Lithium Duct Area = Power/($1.44 * 10^9$ J/m^3)/(3 m/s)

This leads to the following table:

Table 2: Lithium Flow Parameters

Reactor Power	Volumetric Flowrate of Lithium	Lithium Duct Total Cross-section Area	Inner Diameter of Circular Pipe Duct
0.3 MW	$2.08*10^{-4}$ m^3/sec	0.69 cm^2	0.94 cm
1.0 MW	$6.94*10^{-4}$ m^3/sec	2.31 cm^2	1.71 cm
10.0 MW	$6.94*10^{-3}$ m^3/sec	23.1 cm^2	5.42 cm

Assuming that lithium flow within the nuclear reactor occurs in ducts formed between parallel tungsten plates separated 2 millimeters apart, the Reynolds number of the flow is 23077, which is well into the turbulent regime. Moody's diagram[(7)] gives a friction factor of 0.025. This allows calculation of the lithium pressure drop needed to pump through the reactor. If the flow path length between the parallel plates is 10 cm, the reactor's pressure drop will be only 1406 Pa = 0.2 psi.

The heat transfer coefficient within the reactor can be simply estimated by assuming slug-flow conditions in the liquid metal, as suggested by Eckert and Drake[(6)]. The resulting coefficient is h= $3*10^5$ W/m^2-°C. If we allow a 100°C maximum temperature drop between tungsten and lithium, then the heat flux must be limited to

Qmax = $3*10^7$ W/m^2 (= 3 kW/cm^2)

For a 10 MW reactor, a rectangular core measuring about 10 cm X 10 cm X 10 cm would provide adequate heat transfer area to limit heat flux to this level. The core volume would be approximately 25 % liquid lithium, 25% tungsten, and 50 % HEU uranium almost pure in the fissionable U^{235} isotope. Lower power reactors could be more compact.

NEUTRON MODERATOR AND REFLECTOR

The Li^7 isotope is light enough to serve as a neutron moderator, but its cross section for neutron scattering is so small that an excessive quantity of lithium would be needed. Beryllium has a much larger scattering cross section, and could be used as a moderator. But hydrogen is a far better neutron moderator than any other element, which is one reason that water is used so often in reactor designs. Natural lithium reacts with hydrogen to form the hydride, LiH, with average molecular weight 7.95, and which remains a solid with density 0.78 g/cm^3 at temperatures below its 688°C (961°K) melting point temperature. Lithium hydride's hydrogen density can be directly calculated from these values, and is found to be 88% of the hydrogen density in water. So Li^7H can be expected to be a superior moderator at the high temperatures involved in mobile nuclear reactors for Mars.

To approximately estimate the effect of using lithium hydride, the "migration length" value listed for water in Glasstone & Sesonske's Nuclear Engineering text[8] was divided by 0.88 to yield M=6.7 cm. The "rules of thumb" recommended there state that a neutron reflector with a thickness of two migration lengths behaves essentially the same as an infinitely thick reflector, and that the corresponding reflector savings in the radius of the critical mass is reduced by one migration length below a nonreflected reactor.

NEUTRON ABSORBING SHIELD

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The boron-10 isotope has a very large cross-section for absorbing thermalized neutrons, 3838 barns=$3838*10^{-28}$ m^2.[9] Multiplying by its atomic number density N=12.81 * 10^{28} atoms/m^3 gives the macroscopic cross section S=49164/m. So a 1 cm thick layer of B^{10} attenuates thermalized neutrons by a factor of exp(-491.64)= $3*10^{-214}$, which essentially eliminates them. Incompletely thermalized neutrons are attenuated less, but may be reduced dramatically by a 1-cm thick B^{10} shield.

GAMMA RAY SHIELD

Thus, the radius of the reactor and its surrounding moderator/reflector shell and a B^{10} neutron absorbing shell may be about 20 cm for a 10 MW reactor, and slightly smaller for lower power level reactors.

For a 10 MW reactor with maximal equilibrium accumulated fission products, the calculated gamma ray dose without attenuation at a location 3 m from the reactor's center would be 63000 Sv/hr. To be in conformance with USA rules for continuous occupation by radiation workers, the gamma rays in occupied regions should be attenuated to not exceed 5.7 * 10^{-6} Sv/hr. If a location 3 m from the 10 MW reactor center is to be occupied, gamma rays should therefore be attenuated by a factor of $9*10^{-11}$.

A numerical analysis using the tabulated fission gamma energy spectrum, energy absorption coefficients for different photon energies in tungsten, and photon "buildup factors" to estimate the effects of collisions on photon energy [9], leads to the conclusion that the attenuation factor of $9*10^{-11}$ adequate for a 10 MW reactor is provided by a 30 cm thick layer of tungsten. This ignores the substantial gamma ray shielding effects of tungsten and uranium in the reactor core where most of the gamma rays are produced. And it ignores the fact that mechanical engine component could be placed between the reactor and occupied areas, providing some of the shielding.

If all 3 m distant locations in all directions surrounding the reactor were to be continuously occupied, it would make sense to provide the gamma ray shield as a spherical shell from R=20 cm to R=50 cm. The volume of that shell is easily calculated to be 0.49 m^3, and with tungsten's 19300 kg/m^3 density the tungsten mass would be 9.46 metric tons.

If the tungsten shell thickness were reduced to 25 cm, its mass would be reduced to 6.7 tons. However, the radiation dose at 3 m from the reactor's center would reach the allowable annual limit in 187 hours. If the tungsten shell thickness were reduced to 20 cm, the shell mass would be reduced to 4.5 tons. However, the radiation dose at 3 m from the reactor center would reach the allowable annual limit in 4 hours.

The design of long range mobile vehicles should place the nuclear reactor external to the crew compartment, preferably in the rear. Then only one side of the reactor needs to be well shielded. Thinner reactor shielding could be used on the reactor sides where a suited astronaut might only rarely approach during an EVA, and on the top, where an astronaut might never venture at all.

By using these strategies, it should be possible to keep the total mass of a mobile engine including a 10 MW nuclear reactor to about 5 tons. Lower power reactors could have a slightly reduced mass.

AIRCRAFT CHARACTERISTICS AND REQUIREMENTS

The interest in using aircraft on Mars derives from the same advantages that motivate the use of aircraft on Earth. Aircraft can provide long distance mobility at high speed without excessive energy consumption. As stated in High Speed Wing Theory by R.T.Jones and D. Cohen, [11]

> "The chief advantage of air transportation over other forms of travel is the great speed that can be achieved with a relatively moderate cost in terms of energy, or fuel expended per mile of flight. An airplane of efficient aerodynamic form may have a drag less than one twentieth of its weight. The energy required in steady flight is therefore less than one twentieth of the weight times the distance flown. …
>
> It is easily seen that the efficiency of the airplane is the result of the favorable aerodynamic properties of the wing. If one contemplates travel by a rocket, which overcomes gravity and achieves its distance solely by virtue of the kinetic energy imparted at the beginning of the motion, then it is found that the energy

requirement is much greater—of the order of the whole weight time the distance. The energy expenditure of a wingless rocket or projectile thus corresponds to a lift-drag ratio of about one, a figure that can be surpassed easily by almost any form of winged body. …"

On Mars, the aircraft situation differs from the situation on Earth in several respects. The Martian atmosphere is far less dense than Earth's atmosphere, so therefore Martian aircraft designs must be different from Earth designs, requiring much larger areas of aerofoils or wings to develop similar aerodynamic forces. Martian aerofoils or wings can be much thinner since their forces per unit area are considerably less, and they must be much thinner in order to avoid excessive weight. However, their weight is also reduced by the lower Martian gravity.

Traditional airplane designs land or take off while running horizontally on sufficiently smooth surfaces. However, much of Mars' surface is rough and strewn with rocks or boulders, and Mars also has no bodies of liquid water with their characteristically smooth surfaces. With no existing global network of Martian airport runways, the number of locations suitable for a traditional design of airplane to land horizontally would be extremely limited or even non-existent.

For application in Mars exploration missions, it would thus be desirable for aircraft designs to be capable of vertical landings and vertical takeoffs at most locations on Mars' surface. Helicopters are the most well studied machines that provide a capability for vertical landing and takeoff on Earth. If helicopter designs were suitably modified to work on Mars, they could provide the most useful capability for long-range mobility.

Dry Earth air at 0°C has a measured speed of sound of 331.5 m/s (742 mph) whereas carbon dioxide, the main constituent of Mars' atmosphere, has a speed of sound at the same temperature of 259 m/s (579 mph). The speed of sound in colder Mars air is slower, perhaps approaching 210 m/s (470 mph) during polar winter night.

To avoid the drag increase associated with breaking the sound barrier, the designed rotor tip speed could be limited to 140 m/s (313 mph) while the maximum designed cruising speed limited to 70 m/s (157 mph). Then the maximum rotor tip speed through still air would not exceed 70 m/s + 140 m/s = 210 m/s while cruising.

The lift force is given by the formula, $L=C_L\ 0.5\ r\ V^2 S$, where r represents the air density, V represents the airfoil speed with respect to the air, S represents the planform area of the aerofoil, and C_L represents the lift coefficient of that aerofoil geometry. For most aerofoil shapes, C_L varies in rough proportion to the angle of attack, reaching a maximum lift value typically near 1.0 at an angle several degrees from the zero lift orientation.[10] Using r = 0.017 kg/m^3 as a typical Martian air density, and using V=110 m/s as an average airspeed of the helicopter's rotor, this gives L_{MAX}= (200 Pa) S.

Thus for instance, to hover above Mars' surface with a mass of 10 metric tons, a helicopter would need a total aerofoil area of at least S=(10000 kg)(3.7 m/sec^2)/(200 Pa) = 185 m^2. If two 10-meter radius rotor disks were used, slightly less than 30% of the area of the two disks would need to be aerofoils. This is a far greater solid fraction than typical of Earth's helicopters, but it appears possible by using multiple blades per rotor. If the aerofoil Lift/Drag ratio were as low as 10, the engine would need to supply (3700 N)(110 m/s)= 0.4 MW (536 hp) of mechanical power just to maintain hovering. However, the engine must also have additional power for takeoff, landing, cruising, and furthermore should have reserve margins for emergency conditions such as high winds. Probably several thousand horsepower is more appropriate, using a 10 MW reactor driving an open Brayton cycle.

At the proposed 70 m/s cruising speed of this example, it would require 21 hours flying time to travel from Mars' equator to either its North or South pole, or equivalently, 84 hours flying time to completely circumnavigate the entire globe. Because of the high-speed travel, the extent of recycling in the vehicle's life support system could probably be reduced to save weight. For long distance flights, the pressurized cabin space should accommodate at least 2 crewmembers to alternate as pilots.

CONCLUSIONS

The engineering design and development of nuclear powered engines and vehicles for long range mobility on Mars should be pursued now. The developed engines, rovers, and aircraft will provide significant value on Mars at a development and delivery cost which is a miniscule fraction of the entire Mars exploration program cost. Without such equipment, long-range mobility for exploring Mars and for exploiting Martian resources would be extremely limited. The mobility limitations would either cause a large increase in Mars program costs, perhaps leading to early termination of a Mars program, or could even inhibit the first manned missions from occurring. To not undertake design and development of nuclear powered vehicles for Mars application would thus be "penny-wise but pound-foolish".

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