Nuclear Thermal Propulsion for Reducing Radiation Exposure on Manned Missions to Mars
22.04 Final Paper
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I. Problem Statement

One of the several primary technological hurdles to overcome in sending humans to Mars is the issue of excessive levels of astronaut radiation exposure. Career radiation limits for astronauts range from 1-4 Sieverts depending on both sex and age.[1] Even limits set by NASA are concerningly high, however, as astronauts' annual exposure limit is already 10 times the exposure limit set for radiation workers (0.50 Sieverts as compared to 0.05 Sieverts).[1] As such, it is critical to keep the total radiation exposure as low as possible for manned trips to Mars in order to preserve both mission safety and long-term astronaut health upon return to Earth.

Career Exposure Limits for NASA Astronauts by Age and Gender*				
Age (years)	25	35	45	55
Male	1.50 Sv	2.50 Sv	3.25 Sv	4.00 Sv
Female	1.00 Sv	1.75 Sv	2.50 Sv	3.00 Sv

The mechanism in which nuclear thermal propulsion (NTP) reduces radiation exposure to astronauts is by enabling a reduction in transit times to and from Mars, thus decreasing background radiation exposure during transit. Nuclear thermal propulsion is more efficient than chemical propulsion, so with the same spacecraft and fuel mass, nuclear thermal rockets (NTR) have higher delta-v (effectively, higher velocity capability) and thus can achieve faster transits. However, NTP systems themselves also produce radiation during operation, somewhat counteracting the radiation-reducing benefits. The NTP radiation can be decreased through adding additional radiation shielding to the NTP system; however, the added mass from the shielding causes transit times to increase, thus also increasing background radiation exposure. For every spacecraft configuration, there is an optimal shield mass that reduces total radiation exposure (the sum of NTP and background sources).

The goal of this paper is to address whether using nuclear thermal propulsion with an optimized shield mass allows a net reduction in total radiation exposure to astronauts on manned Mars missions. In addition, the effect of changing NTP system parameters (e.g., engine efficiency) on this net radiation reduction will be determined, so as to determine what factors matter the most in achieving a net radiation reduction.

II. Background of Nuclear Thermal Propulsion Technology

Research into nuclear thermal power began in the 1950s with Project Rover, a NASA-run program that developed and tested three different nuclear engine variants. Project Rover's success led to the creation of the Nuclear Engine for Rocket Vehicle Application (NERVA) program that ran until the end of 1972, at which point NASA space program budget

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[1] *The Radiation Challenge*, (NASA, 2008), page 7. https://www.nasa.gov/pdf/284273main_Radiation_HS_Mod1.pdf

cuts following the end of the Apollo program caused the cancellation of NERVA. Research into NTP technology has only recently picked up again, albeit with a shaky start; NASA's Project Prometheus to develop nuclear propulsion systems began in 2000 and was cancelled in 2005 due to a restricted budget.[2] In 2011, NASA began work on the Nuclear Cryogenic Propulsion Stage[3], which continued into the ongoing NTP project in 2016.[4]

The successes of each of the Project Rover and NERVA programs set up a solid foundation of NTP scientific and engineering knowledge for more modern NTP research. By the time of cancellation of the prior projects, NASA had achieved a technology readiness level (TRL) of 6 for nuclear thermal propulsion technology.[2] TRL runs on a scale of 1 to 9, where 1 is when basic principles of the technology have been observed and reported, and 9 is when the technology has been successfully proven on multiple real-world space missions. During that era, NASA ran 22 successful engine tests, and developed engines with specific impulses (a measure of the efficiency of the engine, with the units of seconds) of ranging between 850 to 925 seconds, which is on the order of twice as efficient as the best chemical propulsion engines today (450 seconds).[5]

There are a number of arguments in favor of continuing nuclear thermal propulsion research and using the technology for Mars missions. Most revolve around improving mission feasibility, safety, and cost. Mechanisms for this improvement include reducing radiation exposure to astronauts, reducing mission costs by decreasing the mass of fuel that is sent to orbit, making missions safer through achieving faster mission durations and relaxing mass restrictions on the spacecraft, and simplifying missions by reducing or eliminating the need for refueling (thus also making mission cheaper and safer). There are other numerous benefits to be considered, such as the decreased bone loss of astronauts resulting from shorter transit times, and the decreased energy generation requirements when "bimodal" NTP systems are used concurrently to generate power as well as propulsion for the spacecraft. In general, NTP is a maturing technology, so there is significant room for future growth and increased efficiency. Contrast this growth potential with that of chemical propulsion, which is already a mature technology, and which will likely see little future improvement.

Of all the above factors in favor of using nuclear propulsion for manned Mars missions, this paper only considers the application of NTP to reducing radiation exposure to astronauts.

III. Calculations

A brief overview of the calculation methodology is as follows:

Based on the fixed mass of the spaceship that needs to arrive at Mars, the initial total mass of the spacecraft in low Earth orbit, and the efficiency of the engine, use the rocket

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[2] *Nuclear Rockets: To Mars and Beyond*, (Los Alamos National Laboratory, 2011). https://www.lanl.gov/science/NSS/issue1_2011/story4full.shtml

[3] *The Nuclear Cryogenic Propulsion Stage*, (NASA, 2014), page 2. https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20140012915.pdf

[4] *NASA's Nuclear Thermal Propulsion (NPT) Project*, (NASA, 2018), page 16. https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20180006514.pdf

[5] Specific impulse is measured in seconds, which is dimensionally equivalent to thrust divided by the mass flow rate of the engine. If the engine outputs more thrust with the same mass flow rate, it more efficiently uses its propellant.

equation to calculate the delta-v (change in velocity) that the spacecraft is capable of. Based on that delta-v capability, find the duration of the transit to Mars, the length of the stay on the surface of Mars until the next transfer opportunity, and the duration of the transit back to Earth from Mars. Compute the propulsion radiation exposure by finding the burn time of the engine and multiplying it by the engine radiation exposure rate of the engine. Compute the background radiation exposure by multiplying together transit/surface durations and transit/surface radiation exposure rates.

Perform this calculation on the spacecraft a multitude of times, each time varying the shielding mass by a specified increment in order to find the optimal shielding mass that reduces the total radiation exposure.

The following sections will address (a) the assumptions and limitations of these calculations, (b) the derivation of the figure for the total radiation exposure in the baseline chemical propulsion mission, (c) the orbital calculations for the NTP mission, and (d) the radiation calculations for the NTP mission.

a. Assumptions and Limitations

We must make a number of approximations in order to simplify the problem into a tractable format, as space mission architecture design is a complicated field with numerous many-variable optimization problems, many of which are are outside the scope of this paper. This section discusses preliminary assumptions, justifications for those assumptions, and explanations of the resulting limitations of the model.

Preliminary assumptions related to the calculation of radiation exposure are as follows:

Average out solar flare events. Assume a constant radiation rate of 1.84 mSv/day in space and 0.64 mSv/day on the surface of Mars, as measured by the Radiation Assessment Detector module on the Mars Curiosity rover, which provides us with accurate data for expected exposure rates both during the trip to Mars and during the stay on the Martian surface.[6] In reality, some Mars missions will be able to entirely avoid solar flare incidents, and therefore will only receive the continuous baseline exposure rate, while other missions will be unlucky and get hit by several solar flare incidents. On average, the exposure rate should be consistent, however; besides, such solar activity events will be consistent between chemical and nuclear propulsion missions, and thus are unimportant in analysis of this problem.

Average out solar cycle minimum/maximums. During the 11-year-long solar cycle, background radiation exposure rate experiences variation. There is only some reason to believe that nuclear propulsion would be able to take advantage of solar cycle minimums by launching during those windows. By having the shorter mission durations that correspond with nuclear propulsion systems, the launch and return windows would fit more “snugly” into the solar cycle minimum. This effect is insignificant, however, as the mission times between NTP and chemical systems only vary by ~100 days for the mission architectures assumed in this paper (long-stay

[6] *Space Weather & the Radiation Environment at Mars: Energetic Particle Measurements with MSL RAD*, (Southwest Research Institute, 2015), page 10, 17.

minimum energy and long-stay fast-transit missions). In addition, the uncertainties involved in predicting solar cycle behavior renders taking advantage of solar cycle minimums unsafe.

Assume astronauts receive no radiation protection from background sources.

Assume engine radiation shielding contributes no shielding protection to background radiation as well. This is a reasonable approximation, because the propulsion radiation shielding does not surround the astronauts, but merely lies between the astronauts and the propulsion system and thus only covers a small fraction of their line of sight to space. This factor does result in a slight underestimate of the ideal amount of shielding to use, however.

Assume engine radiation shielding mass is purely the mass of the shielding material itself, i.e., assume that the mass of the structure supporting and containing the shielding material is negligible. Note that in this model, 1 kg of added water for shielding contributes 1 kg to the mass of the spacecraft. In reality, using water as a shielding material may permit less strict requirements on the life support system, as the necessity for high water reclaiming capability would be relaxed if the shielding water could slowly be drained for use by astronauts (for example, for showering, drinking, growing crops, etc.), and then replenished by mining ice at Mars. A lower-performing life support system would weigh less than a higher-performing life support system. Therefore, using water to shield astronauts from radiation may actually add less mass to the spacecraft than the mass of the water itself, since mass would be saved from the life support system. The effect of this assumption on the model is unclear.

For simplicity, only consider gamma radiation from the engine; ignore neutron radiation. Neutron radiation will be largely be blocked by the LH_2 propellant tanks that sit between the engine and the astronauts.[7] As a result, the model will underestimate NTP exposure and underestimate the appropriate shielding mass.

Assume the engine produces negligible radiation upon startup and once it is turned off. In reality, the engine continues to emit radiation on the order of a few Rad/hour in the weeks and months following engine shut-off.[8] Such radiation is a small fraction of the radiation produced during engine operation, however. As a result, the model will underestimate NTP exposure and underestimate the appropriate shielding mass.

Preliminary assumptions relating to orbital calculations are as follows:

Only compute delta-v capabilities for the transit to Mars from Earth; assume the same delta-v capability for the return trip. This simplification is an acceptable approximation in the case that creating LH_2 propellant on Mars is feasible and the transit spacecraft is “parked” in Mars orbit during the astronaut’s stay on Mars.

Orbital calculations and trip-time optimization problems are complicated. Use pre-generated data points from online resources, and simply interpolate between data points by creating best-fit polynomials. This method introduces negligible error as long as data is only interpolated from, not extrapolated from. Note that later on in this paper, we use orbital trajectories for the 2024-2026 timeframe; this is a reasonable exception to not using extrapolation, because planets’ orbits years from now are highly predictable.

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[7] *Shielding Development for Nuclear Thermal Propulsion*, (NASA, 2015), page 3. https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20150006884.pdf

[8] *NERVA*, (Aerojet Nuclear Systems Co., 1970), page 75. https://fas.org/nuke/space/nerva-spec.pdf

Assume no staging mechanisms exist for the spacecraft once it is in low Earth orbit. As a result, the model will underestimate NTP efficiency compared to real-world missions, and predict a lesser reduction in total radiation exposure than is actually true.

Assume system mass does not vary with total mission time (in particular, assume a constant life support system mass). Required resources do not vary significantly with moderate variances in total mission time. Water is recyclable and ice can be potentially mined on Mars to restore water supplies. Food can be grown on Mars, and otherwise represents a negligible fraction of the total mass anyways. Required redundancies or reliability of the life support system may alter its mass for different mission durations, but ignore this potential effect.

Assume high-speed aerobraking is feasible no matter the Mars or Earth atmospheric entry velocities. In reality, aerobraking maneuvers are highly velocity-limited, and the spacecraft must expend more fuel to slow itself down before entering the atmosphere and using drag to slow itself the rest of the way. Calculating the optimization solution numerically, however, requires recursive computation of the spacecraft's transit duration and delta-v capabilities. Atmospheric entry velocities increase with decreased transit times, which in turn requires more delta-v to be expended before aerobraking, which results in less delta-v available for transit, which results in increased transit times, which results in decreased atmospheric entry velocity, and the cycle repeats. As a result, the model will overestimate NTP efficiency and thus predict a greater reduction in total radiation exposure than is actually true.

A further explanation of limitations to the accuracy of engine efficiency analysis

There are two primary limitations to the analysis of the effect that engine efficiency has on reducing net radiation exposure: (1) the assumed orbital path of the spacecraft, and (2) the disregard of velocity-limited aerobraking capabilities.

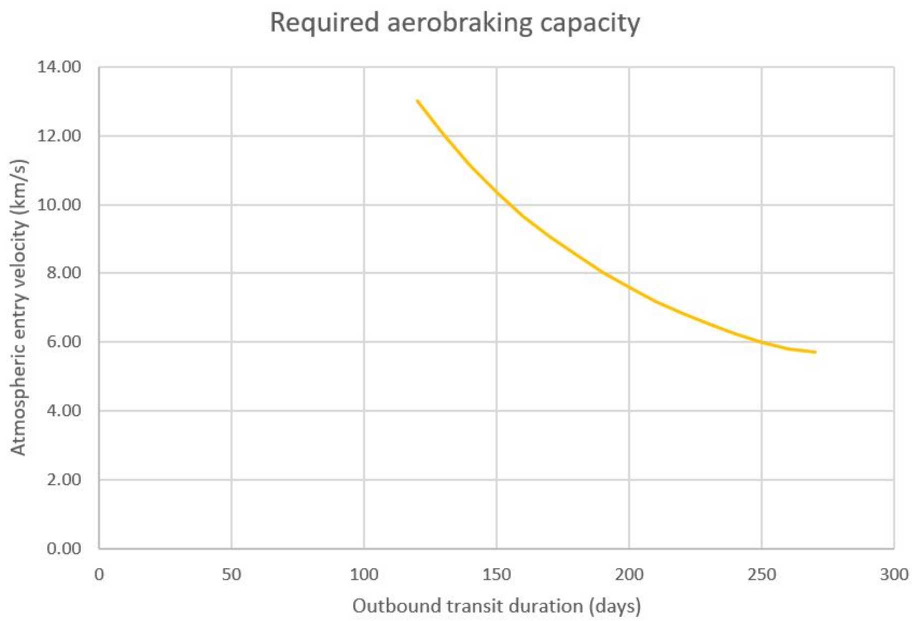
(1) The assumed orbital path of the spacecraft

In order to provide the best "apples-to-apples" comparison of nuclear and chemical rockets, this paper assumes a long-stay fast-transit trajectory for nuclear thermal rockets, and a long-stay minimum energy trajectory for chemical rockets. Nuclear thermal rockets are capable of any kind of trajectory however, including minimum energy, fast-transit, and short-stay missions (as depicted in the following section), while chemical rockets are only capable of long-stay minimum energy missions. Choosing to use NTP for these other trajectories may allow for further possible radiation reduction. Short-stay missions, for example, would halve the duration of the mission and thus would approximately halve the total radiation dose. Using NTP on long-stay minimum energy missions would give the spacecraft a remarkable surplus of delta-v capabilities, increasing the potential payload. This payload could even include radiation shielding for background radiation, which may significantly decrease overall radiation even if the mission times are increased. So for this section, keep in mind that we are only analyzing long-stay missions.

(2) The disregard of velocity-limited aerobraking capabilities

Using excess delta-v capabilities to further decrease transit times comes at a cost, which has been entirely neglected in this analysis. Whenever transit times are decreased, the spacecraft correspondingly moves faster relative to its destination, and thus arrives at its destination with a higher velocity as well. Normally, the spacecraft does not need to use its fuel to slow down from this increased velocity, because it simply uses drag in the atmosphere of the destination planet to slow itself down for landing. This aerobraking capability is not available in unlimited capacity, however; if the spacecraft performs this maneuver at excessively high velocities, it will begin to overheat, and potentially disintegrate within the planet's atmosphere before it is able to land.

The following graph depicts this tradeoff:



If the spacecraft can achieve 13 km/s aerobraking capabilities, this effect can be ignored. In reality, for landing massive payloads on Mars, achieving this aerobraking capability may prove exceedingly challenging. For reference, the Mars Curiosity Rover entered the Martian atmosphere at approximately 5.5 km/s.[9]

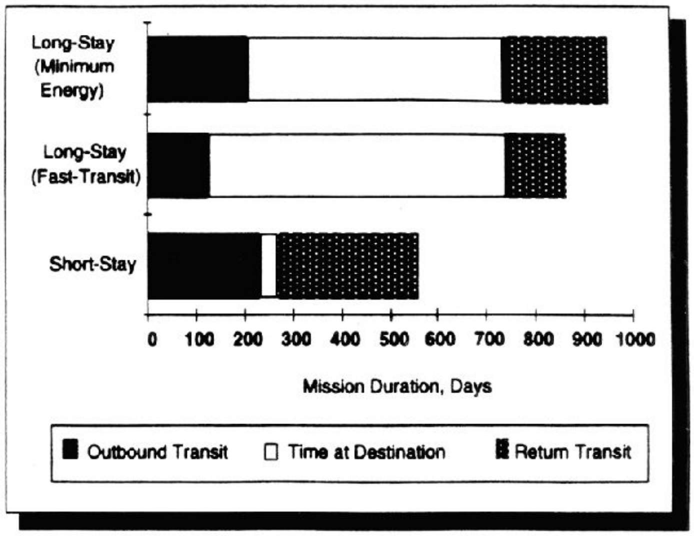
Thus for increased engine efficiencies, where the primary mechanism for decreasing total radiation exposure is decreasing transit times, the arrival velocity will correspondingly increase. For low arrival velocities, no additional considerations are required; the spacecraft can simply perform a more aggressive aerobraking maneuver. When the spacecraft reaches its aerobraking capacity, however, this effect can be dealt with in one of two ways. Either the spacecraft can save excess fuel to slow itself down just before entering the destination planet's atmosphere, or the spacecraft can come equipped with a better (i.e. more massive) heat shield in order to withstand more heat upon aerobraking. Each of these methods reduces the delta-v of the spacecraft, however, and thus also reduces the efficacy of its transit duration reduction.

The following sections detail calculation methodologies and equations used within the excel file that performs the calculations.

[9] *Entry, Descent, and Landing*, (JPL, 2012). https://mars.nasa.gov/msl/mission/timeline/edl/

b. Baseline Chemical Propulsion Mission Architecture Calculations

There are three broad categories of manned Mars missions: long-stay minimum-energy; long-stay fast-transit; and short-stay missions. A comparison of these categories are shown below:[10]



Short-stay missions are not considered within this paper due to the limited time at the Mars surface that they allow. Fast-transit missions are not achievable by current chemical propulsion technology, as they have excessively high delta-v requirements. Within this paper, we assume that NTP missions will follow long-stay fast-transit trajectories, and chemical propulsion missions will follow long-stay minimum energy trajectories.

For the baseline chemical mission architecture, assume an outbound transit duration of 250 Earth days, a surface duration of 500 days, and an return transit duration of 250 days. This results in a total radiation dose of 1.24 Sieverts.[10] All NTP mission radiation levels will be compared to this value.

c. Orbital calculations

The goal of this section is to explain the derivation of transit times, surface stay times, and engine burn times. These values will be multiplied with their respective radiation exposure rates, as computed in the following section, to determine the total radiation exposure from each source.

Data points are taken from orbital calculations for the 2024-2026 Mars mission window (years are chosen arbitrarily). Orbital trajectory data comes from MARS, the International Journal of Mars Science and Exploration.[11] The following graphs, created from this data, relate spacecraft delta-v capabilities to the corresponding transit times for these launch windows.

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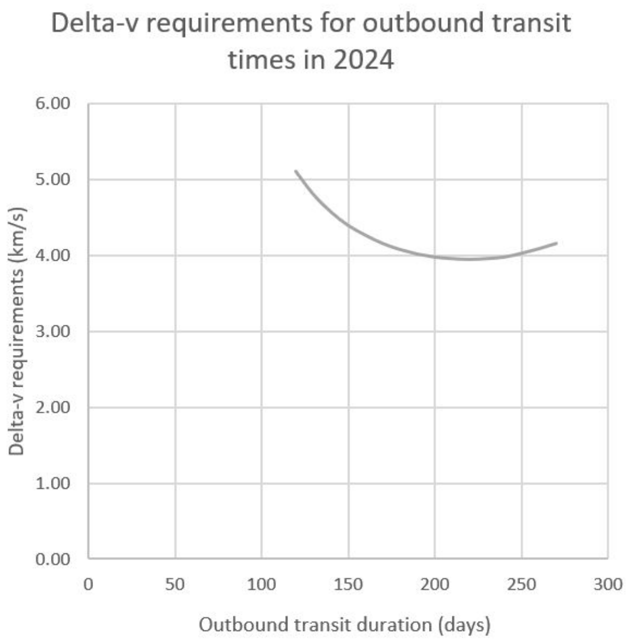
[10] *A Crewed Mission to Mars*, (NASA, 2015). https://nssdc.gsfc.nasa.gov/planetary/mars/marsprof.html
[11] *Mission Design Options for Human Mars Missions*, (Mars, 2007), page 16, 17. http://marsjournal.org/contents/2007/0002/files/wooster_mars_2007_0002.pdf



Transit durations from this data source are only reported in 10-day increments, so in doing the calculations within this section, we approximate the curve using a best-fit polynomial and input the delta-v capability of the spacecraft to get the output of the lowest possible transit duration. This method allows for interpolation between data points.

Spacecraft delta-v is calculated by using the rocket equation, assuming an initial mass of 336.5 metric tons and a final baseline mass of 203.1 metric tons (baseline mass meaning mass without taking radiation shielding into account).

$$\Delta v = I_{sp} * g_0 * ln(\frac{m_0}{m_f})$$

Several iterations are performed, where the radiation shielding is increased in 100 kg increments until the spacecraft has insufficient delta-v to reach Mars at all. In this methodology, we assume that the spacecraft has the same delta-v capability on the return trip from Mars as it does when it originally travels there. This assumption is sensible in the scenario that the spacecraft refuels at Mars, either using orbital depots or by generating fuel on the Martian surface.

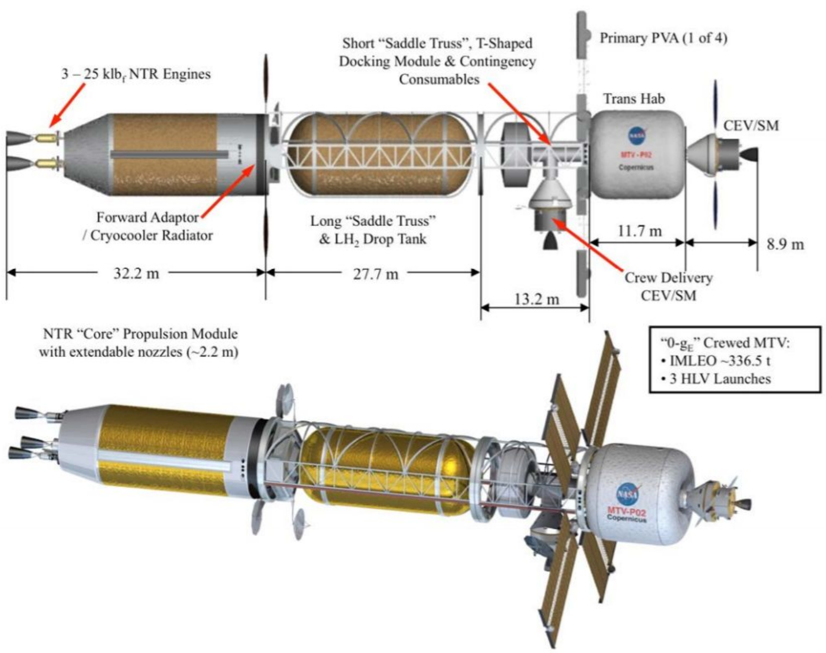
Using the above graphs, the total transit duration times are calculated from the spacecraft delta-v using the best fit polynomials. Surface stay duration for fast-transit missions are approximately equivalent regardless of transit times (650 ± 10 days), so we will simply approximate this value as 650 days.

Engine burn time is computed as follows:

$$burn\ time = \Delta m * g_0 * I_{sp} / F$$

Where delta-m is the change in mass of the rocket (the consumed fuel), g_0 is standard gravity, I_{sp} is the efficiency of the engine (900 sec) and F is the thrust.

The mass, spacecraft dimensions, and engine efficiency figures from this section are based on NASA's Design Reference Architecture (DRA) Study 5.0, as depicted below.[12] Unlike the DRA, these calculations assume one nuclear engine rather than three.



d. Radiation Calculations

The goal of this section is to explain the derivation of the engine radiation rate, and the background radiation rate on the Martian surface and during transit. These values are multiplied by their respective exposure times which were computed in the above section in order to obtain the total radiation exposure estimate.

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[12] *Nuclear Thermal Propulsion (NTP): A Proven Growth Technology for Human NEO / Mars Exploration Missions*, (NASA, 2012), page 12. https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20120003776.pdf

Engine radiation exposure

The cumulative radiation rate of a potential nuclear thermal propulsion system during operation is estimated to be on the order of 10^9 Rad/hour (summed over all directions).[13] To calculate the fraction of this radiation that is transmitted to onboard astronauts, we first calculate the exposure rate to an astronaut that is a certain distance away, assuming there is no shielding. This equation follows the inverse square law, as an astronaut that is twice as far away will experience a quarter the radiation from the engine.

$$total\ radiation\ rate\ from\ engine\ *\ \frac{human\ cross{-}sectional\ area}{4\pi r^2} = human\ exposure\ rate\ without\ shielding$$

A typical person's cross-sectional area is ~1 m^2 from the front, and ~0.5 m^2 from the top. Here we conservatively approximate this value as 1 m^2.

To calculate the human exposure rate with shielding, we multiply the non-shielded rate by the fraction of radiation that manages to pass through the shielding. This fraction is determined by both the attenuation factor of the shield material and the thickness of the shield.

$$human\ exposure\ rate\ without\ shielding * fraction\ passed\ through\ shielding = human\ exposure\ rate\ with\ shielding$$
$$human\ exposure\ rate\ without\ shielding\ * e^{-attenuation\ of\ shield\ material\ *\ shield\ thickness} = human\ exposure\ rate\ with\ shielding$$

Water is one possible shielding material for gamma radiation. Water has an attenuation of 0.1 cm^2/g, density of 1 g/cm^3, and therefore a linear attenuation of 0.1 cm^{-1}.[14] The shield is assumed to be a cylinder in-line with the spacecraft, placed between the astronauts and the nuclear propulsion system, with no mass contribution from the water containment (i.e. the water container has zero mass).

$$area * shield\ thickness * shield\ density\ =\ mass\ of\ shield$$
$$shield\ thickness\ =\ \frac{mass\ of\ shield}{\pi r^2 * shield\ density}$$

Once the radiation rate from the engine is calculated, the total engine radiation exposure is simply the radiation rate multiplied by how long the nuclear engine is in operation (the "burn time"). The burn time is calculated using the amount of fuel burned (Δm), the efficiency of the engine (I_{sp}), a constant factor (g_0, "standard gravity"), and the thrust of the engine.

$$\frac{\Delta m * I_{sp} * g_0}{thrust} = burn\ time$$
$$burn\ time * radiation\ rate\ =\ radiation\ exposure\ from\ the\ engine$$

[13] *Nuclear Thermal Propulsion*, (NASA, 2019). https://www.sbir.gov/sbirsearch/detail/1547743

[14] *Gamma Ray Attenuation of Common Shielding Materials*, (PG Research Foundation, 2018), page 7. https://www.eichrom.com/wp-content/uploads/2018/01/gamma-ray-attenuation-white-paper-by-d.m.-rev-4.pdf

Background radiation exposure

The total radiation from background radiation exposure is a more straightforward calculation:

$$total\ background\ exposure\ =\ travel\ duration * space\ exposure\ rate\ +\ surface\ duration * Mars\ exposure\ rate$$

The Radiation Assessment Detector module on the Mars Curiosity rover provides us with accurate data for expected exposure rates both during the trip to Mars and on Mars itself. Astronauts will be exposed to an average of 1.84 mSv/day during transit and 0.64 mSv/day during their stay on the Martian surface.[15]

IV. Results and Analysis

There are several flexible parameters of NTP systems that are either not yet precisely determined for propulsion systems that would be used for actual Mars missions, or are otherwise independent parameters that can be individually controlled. These parameters include the engine's radiation rate, specific impulse (efficiency), mass, thrust, thrust-to-weight ratio, dimensions (in particular, its radius), and distance to the crew. Several of these have negligible on the overall radiation exposure to astronauts. The rest will be analyzed within this section.

The parameters that will be ignored in this analysis are engine mass, thrust-to-weight ratio, and thrust. The engine's mass is a fairly minimal proportion of the overall spacecraft mass, and thus does not significantly influence calculations. The same is therefore true of its thrust-to-weight ratio, so both of these factors are ignored. Engine thrust only affects the duration that the engine needs to fire (its "burn time"). The length of the burn time is of course directly related to radiation exposure from the engine, but since engine thrust and radiation rate are closely (albeit not necessarily perfectly) correlated, and engine thrust and burn time are perfectly inversely correlated, changing the thrust of the engine does not (in theory) affect the cumulative radiation exposure. The exact dimensions of the NTP system initially may appear to be an unimportant factor, but as the following analysis will demonstrate, engine diameter has a critical impact on the the required shield mass, and thus the transit duration as well.

Since the remaining parameters have uncertain values, we will try to ascertain their values as best as possible, and then determine how much varying each of those values in either direction (while holding the others fixed) influences the amount of radiation exposure the astronauts receive. This will also determine which factors engineers should focus on improving the most when developing a nuclear powered rocket.

[15] *Space Weather & the Radiation Environment at Mars*, (Southwest Research Institute, 2015), page 14, 17.
https://indico.cern.ch/event/390724/contributions/1824626/attachments/1176180/1700529/05_-_Hassler_AMS-02_RAD_talk_23oct15_FINAL.pdf

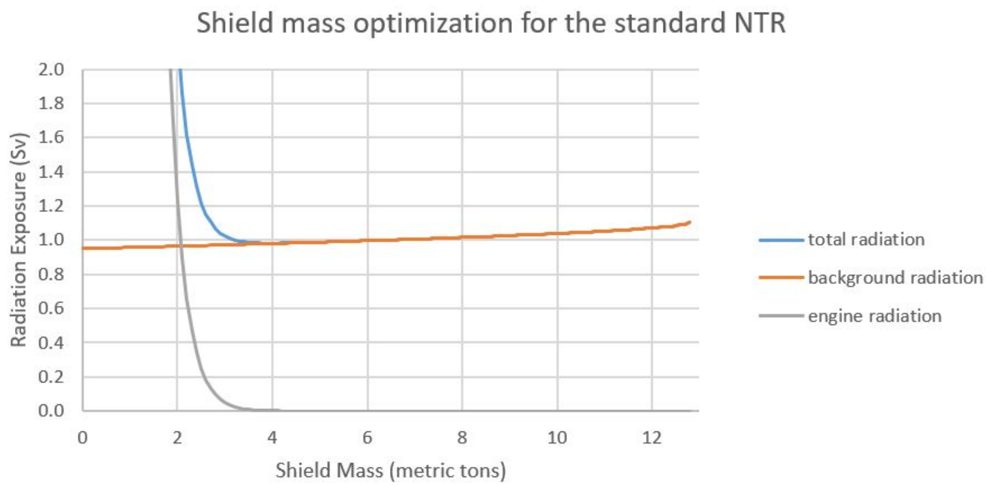
Initial values:

Distance between engine and crew capsule: 80 meters[16]

Specific impulse: 900 seconds[16]

Engine radiation rate: 10^9 Rad/hour[17]

Engine radius: 1 meter[18]



For the standard nuclear thermal rocket, the optimal shield mass is 3.9 metric tons, which gives the astronauts a total radiation exposure of 0.983 Sieverts. Notably, this is just below the career limit of 1 Sievert. From the graph above, we can see that this minimum is achieved when the engine radiation drops essentially to 0. Further increasing the shield mass simply increases the transit duration of the trip, thus increasing background radiation exposure, negating the effect of further decreasing the engine radiation exposure. The shield mass cuts off at 12.9 metric tons, because at more than 12.9 metric tons, the NTR has insufficient delta-v to reach Mars assuming it takes a fast-transit trajectory. The same holds true for the following graphs in this section -- shield masses exceeding 12.9 metric tons are unachievable.

In the following figures, we will simply plot the total radiation dose rather than also continuing to plot the background and engine radiation doses.

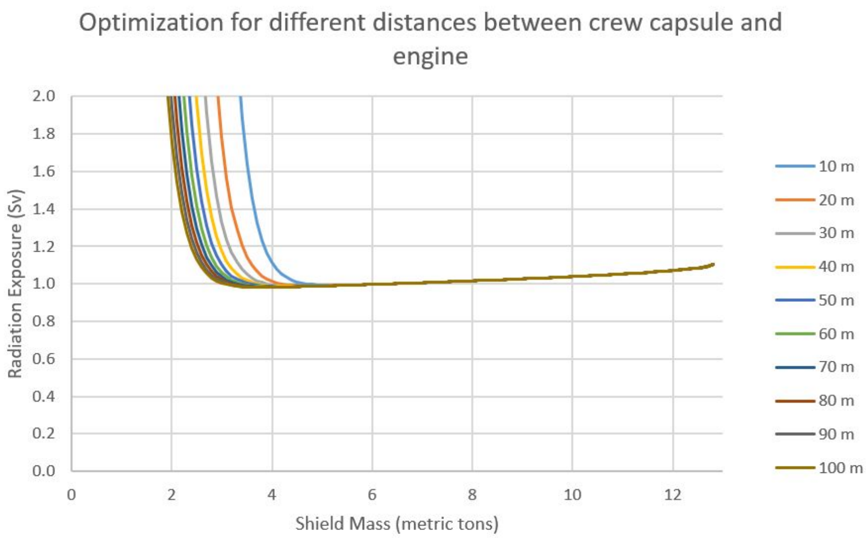
Distance between the engine and crew capsule

Varying the distance between the engine and the crew capsule changes the radiation exposure, because when astronauts are further away from the engine, they receive a lesser exposure due to the inverse square law. Varying this distance from 10 to 100 meters in the following plot displays the effect this has on the total radiation dose.

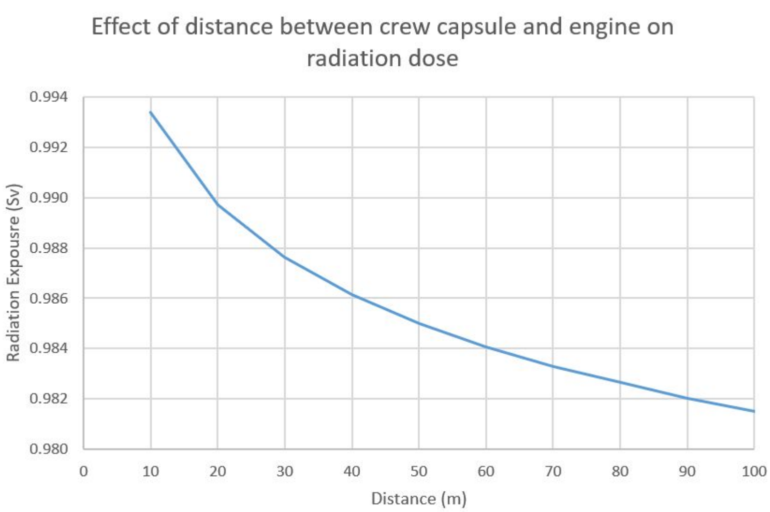
[16] *Nuclear Thermal Propulsion (NTP): A Proven Growth Technology for Human NEO / Mars Exploration Missions*, (NASA, 2012), page 1. https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/20120003776.pdf

[17] *Nuclear Thermal Propulsion*, (NASA, 2019). https://www.sbir.gov/sbirsearch/detail/1547743

[18] *Rover/NERVA-Derived Near-Term Nuclear Propulsion*, (NASA, 1992) https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19930017729.pdf



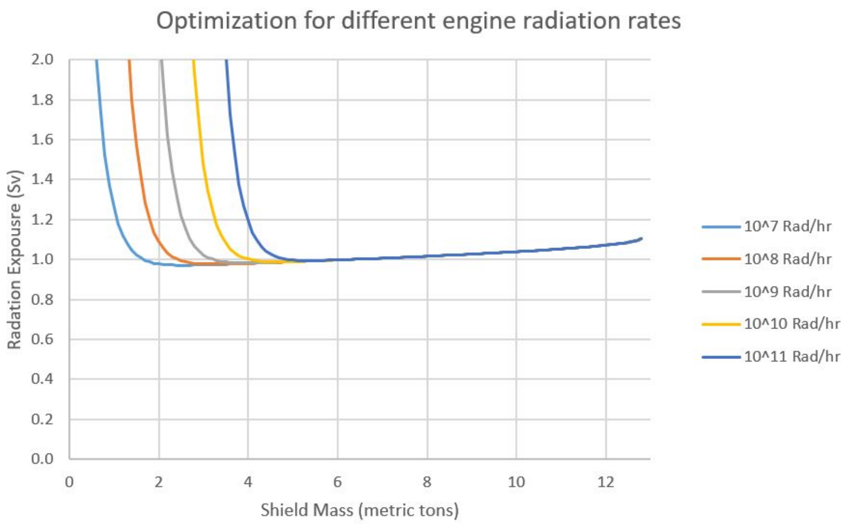
For larger distances, it takes less radiation shielding to achieve the minimum possible total radiation dose. Plotting the minimums for each distances from this graph gives us the following graph:



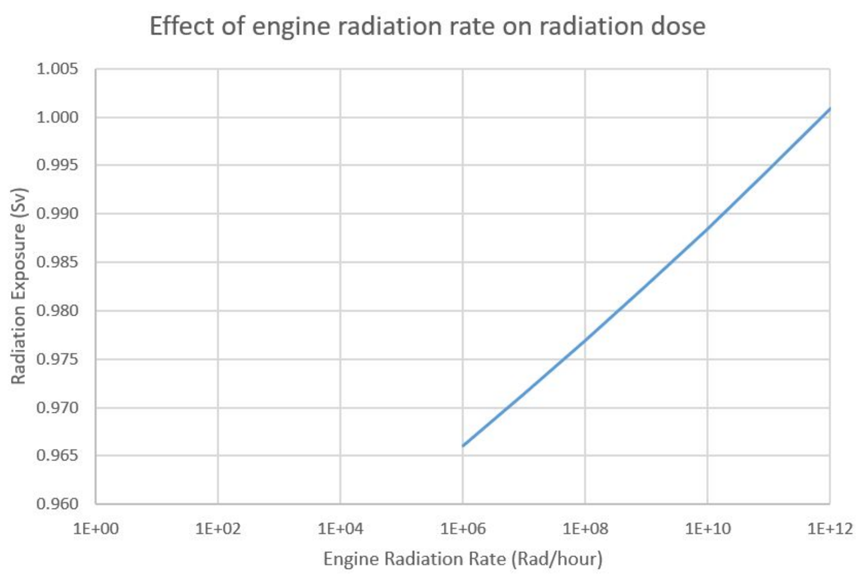
The effect of changing this distance on reducing total exposure is mostly negligible. Increasing the distance from 20 meters to 80 meters, for example, only drops the exposure by 0.5%. Therefore the distance between the crew capsule and the nuclear propulsion engine is a factor that should likely not be considered in the design of a nuclear powered Mars rocket. In fact, while this graph assumes that altering the distance has no effect on the mass of the spacecraft, increasing the distance will actually increase the mass of the spacecraft in reality. Therefore it is possible that increasing this distance will actually increase the radiation dose, since the mass of the spacecraft will also increase. For these reasons, we will discount any effect that altering this distance has on the overall exposure (although a direct comparison of the weight and distance tradeoff have not been calculated).

Engine radiation rate

The following graph shows the effect of changing the engine radiation rate on the total radiation dose. Note that the values of the engine radiation rate (e.g. 10^9 Rad/hr) are for its cumulative output in all directions, not output directly to astronauts.



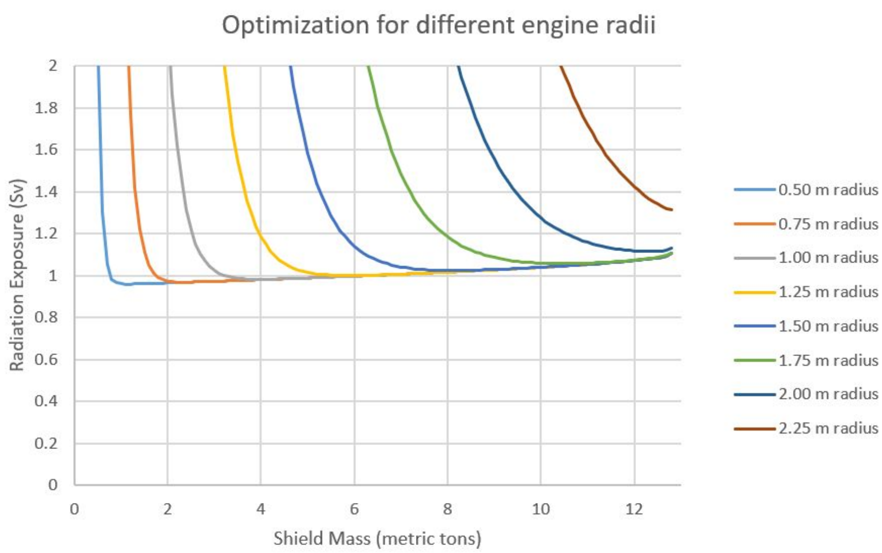
Sensibly, increasing the engine radiation rate causes the spacecraft to require more radiation shielding in order to achieve the minimum possible exposure. Once again plotting the minimums, but now logarithmically, for ease of understanding:



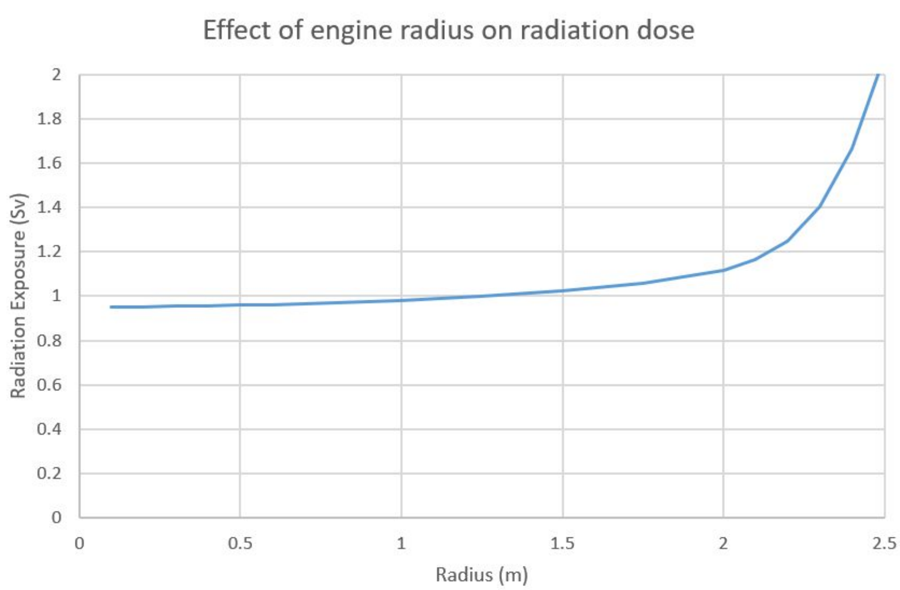
This graph demonstrates that changing the engine radiation rate has a minimal effect on the overall radiation exposure. Altering the assumed rate (10^9 Rad/hr) by an order of magnitude in either direction only changes the overall exposure by only approximately ±0.5%. This change is small because when the optimal shield mass is achieved, the engine radiation exposure comprises only a very small proportion of the overall exposure (i.e. the engine exposure is much smaller than the background exposure). The above change results mostly from the slowed transit as a result of requiring more shield mass.

Engine radius

Altering the radius of the engine also alters the radius of the shield that is required to block its radiation. For this analysis, the required radius of the shield is assumed to be the same as the radius of the engine. The efficacy of the shield is only determined by its depth, however, not its radius, and increasing the shield radius while holding its depth constant proportionally increases the shield mass by the square of the radius. Therefore increased engine radii require much more massive shielding, which in turn increases transit times, and thus also increases background radiation exposure.



Taking the minimums of each curve from this plot gives the following graph:

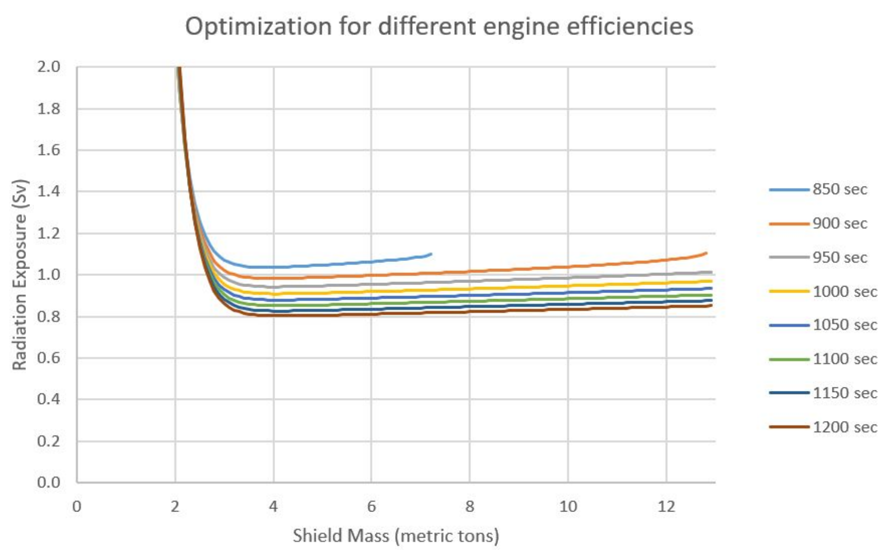


Note that prior nuclear engines from the Project NERVA era ranged in radius from ~0.5 to 5 meters.[19] We can observe from this graph, however, that radii much beyond 2 meters will end up significantly increasing radiation exposure to astronauts due to the mass of shield required. There are two general “modes” of this graph -- the former, from 0 to 2 meters, is where the spacecraft is capable of carrying sufficient shield mass to adequately diminish the engine radiation exposure to acceptable levels. The change in total exposure in this mode primarily results from the change in transit times. For engine radii beyond 2 meters, the spacecraft becomes mass-limited and thus the maximum possible amount of radiation shielding does not reduce the engine radiation exposure to acceptable levels. The total radiation exposure starts to dramatically increase due to engine radiation.

Engine efficiency

Particularly in this portion of the analysis, recall the caveats discussed in the assumptions and limitations section of this paper. Due to restricted aerobraking capabilities, increasing engine efficiency likely has a much lesser effect on reducing radiation exposure than this section would suggest.

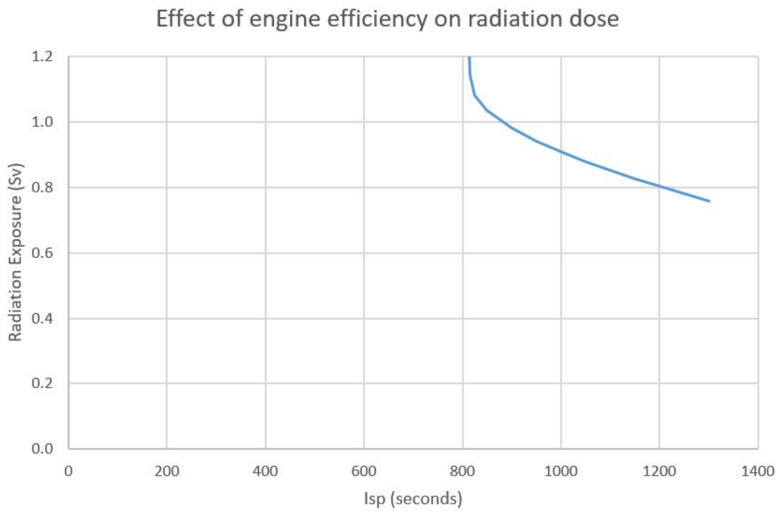
With these caveats in mind, we can consider the following graphs:



The reason the 850 second engine efficiency has a lower cutoff is because the spacecraft’s maximum carrying capacity is reduced (i.e. with a specific impulse of 850 seconds rather than 900 seconds, it can only carry a maximum shield mass of approximately 7 metric tons rather than 13 metric tons). The cutoffs for engines with specific impulses of 950+ seconds are beyond the limits of this graph.

Plotting the minimum radiation values for various engine efficiencies gives the following graph:

[19] *Rover/NERVA-Derived Near-Term Nuclear Propulsion*, (NASA, 1992) https://ntrs.nasa.gov/archive/nasa/casi.ntrs.nasa.gov/19930017729.pdf

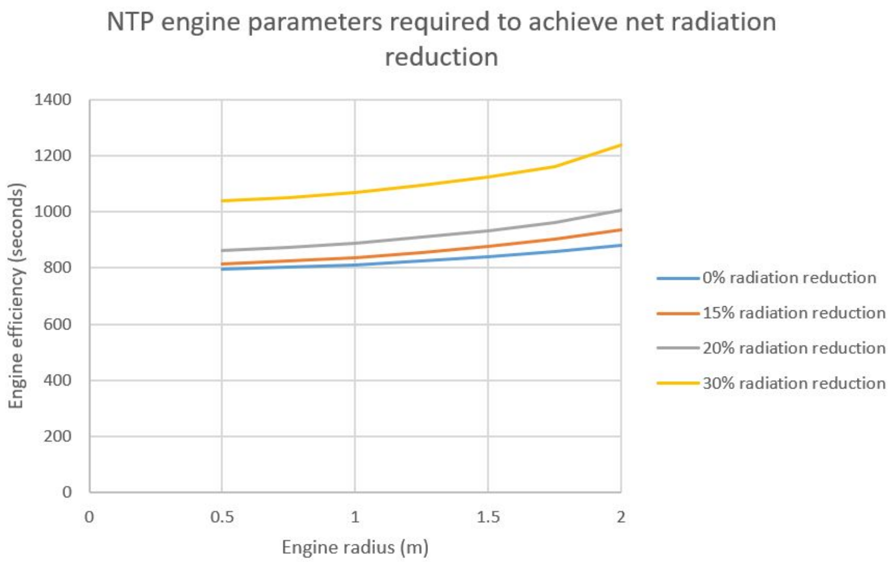


The spike at I_{sp} = 800 seconds results from the fact that in order to have sufficient delta-v to follow fast-transit trajectories at this engine efficiency, the spacecraft will hardly be able to carry any shielding mass at all.

Due to the limitations of this analysis as discussed above (and assuming a fast-transit trajectory with no radiation shielding for background exposure), this graph will likely show a less significant decrease in radiation exposure for higher engine efficiencies. The general trend will still hold that increasing the engine efficiency decreases the total radiation exposure, however.

With these calculations in mind, we can now postulate a hypothetical best and worst case scenario for NTP technology. A reasonable best case scenario is one in which an NTP system has, perhaps, a 0.5 meter radius, 1000 second specific impulse, is placed 100 meters from the crew capsule, and emits radiation at 10^9 Rad/hr. This configuration results in an exposure of 0.89 Sieverts, or a 28% reduction from the baseline chemical propulsion mission of 1.24 Sieverts. A reasonable worst case scenario is one in which an NTP system has perhaps a 1.5 meter radius, 850 second specific impulse, is placed 40 meters from the crew capsule, and still emits radiation at 10^9 Rad/hr. This configuration results in an exposure of 1.20 Sieverts, corresponding to a 3% reduction.

Considering that engine efficiency and engine radius are the two dominant factors in determining total radiation reduction, we can also determine a curve that displays necessary values for each in order to achieve certain percentage reductions compared to the 1.24 Sv baseline.



As shown by these curves, it is significantly more difficult to achieve a 30% reduction in radiation than it is to achieve a 15-20% reduction, and a 15-20% reduction is only marginally more difficult than achieving no reduction at all. Note that a 20% reduction is all that is necessary to bring the baseline radiation below the minimum career limit of 1 Sv.

V. Summary of Findings

With the assumed nuclear thermal rocket parameters, the radiation dose on a manned mission to Mars using NTP technology is 0.98 Sieverts per astronaut. This grants a 21% reduction in radiation from the baseline chemical propulsion mission dose of 1.24 Sieverts per astronaut, and is notably just below the career radiation limit of 1 Sievert per astronaut. Depending on the specific NTP parameters, this reduction may range anywhere from 0% to 30% in the likely worst or best case scenarios.

Certain factors that may be intuitively assumed to significantly alter this cumulative radiation dose end up having a negligible effect. These factors include the distance between the engine and the crew capsule, the radiation rate of the engine, the thrust of the engine, and the thrust-to-weight ratio of the engine.

Other factors end up having a strong effect on the resulting radiation dose; in particular, (1) the efficiency of the engine, and (2) the radius of the engine each significantly change the total cumulative radiation dose during the mission. The mechanism for this radiation reduction for each are as follows: increasing the efficiency of the engine allows faster transits, thereby decreasing background radiation exposure; decreasing the radius of the engine allows a smaller radiation shield to be used to protect astronauts from its exposure, thereby allowing faster transits, and thus also decreasing background radiation exposure. It is worthwhile to note that these two engine parameters are in direct competition with each other, as it is a general trend that in order to increase the efficiency of a nuclear thermal engine, one must also increase its radius.[20] The tension between these two factors and how they should be weighed against each other remains an area for further analysis.

In regards to reducing radiation dose, the best-case scenario for NTP technology is one in which the NTP system is as compact and efficient as possible. The main requirements for this radiation reduction to be realized are (1) for the engine efficiency to be ~850 seconds or above, and (2) for the engine reactor core radius to be ~2 meters or below. Beyond these approximate cutoffs, NTP systems will end up increasing rather than decreasing total radiation exposures.

Although nuclear thermal propulsion will assist manned Mars missions when it comes to the issue of radiation exposure as long as these base requirements are met, it is not necessarily an enabling or necessary technology, as it grants only an approximate 20% reduction in exposure.

[20] *Nuclear Thermal Rocket/Vehicle Design Options for Future NASA Missions to the Moon and Mars*, (NASA, 1993), page 3. https://trajectory.grc.nasa.gov/aboutus/papers/AIAA-93-4170.pdf