İSTANBUL TECHNICAL UNIVERSITY ★ INSTITUTE OF ENERGY

USING NUCLEAR ENERGY IN A SPACECRAFT FOR PROPULSION AND POWER IN A MICROGRAVITY ENVIRONMENT

M.Sc. Thesis by
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Department : Energy Science and Technology

Programme : Energy Science and Technology

JANUARY 2010

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JANUARY 2010

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UZAY ARAÇLARINDA NÜKLEER ENERJİNİN TAHRİK SISTEMLERİ VE GÜÇ ÜRETİMİ İÇİN MİKROGRAVİTE ORTAMINDA KULLANIMI

YÜKSEK LİSANS TEZİ

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OCAK 2010

To the Loving Memory of
Prof. Dr. M. Akif Atalay

FOREWORD

I would like to express my deep appreciation and my gratitude for my advisor Prof. Dr. Murat Aydın in this thesis. His immense help in helping me to plod through this thesis was extraordinary, and this project could never have been completed without his help.

In addition, I also wish to reminisce Prof. Dr. Akif Atalay, who was my first thesis advisor. I dedicate this thesis wholly to his loving memory, as he was a great source of enlightenment to me. My only hope is for this thesis to be known as his legacy, as I devote this thesis fully to him.

December 2009 Uğur GÜVEN

Aerospace Engineer

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ABBREVIATIONS

ESA	**:** European Space Agency
NASA	**:** National Aeronautics and Space Administration
NERVA	**:** Nuclear Engine for Rocket Vehicle Application
NTR	: Nuclear Thermal Rocket
UN	**:** United Nations
RORSAT	**:** Radar Ocean Reconnaissance Satellite

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LIST OF SYMBOLS

C60	**:** Carbon 60
D	**:** Deuterium
g	**:** Gravitational Constant
h	**:** Enthalpy
H	**:** Hydrogen
$\mathbf{H_2}$	**:** Di-atomic Hydrogen
$\mathbf{I_{sp}}$	: Specific Impulse
Nu	: Nusselt Number
k	: Thermal Conductivity
P	: Pressure
Pr	: Prandtl Number
S	: Separation Ratio
T	: Temperature
U	: Stream Gas Velocity
$\mathbf{U^{238}}$	**:** Uranium 238 Isotope
$\mathbf{U^{235}}$	**:** Uranium 235
$\mathbf{UF_6}$	**:** Uranium Hexafluoride
$\mathbf{V_e}$	: Nozzle Exit Velocity
$\boldsymbol{\varepsilon}$	: Nozzle Expansion Ratio
$\boldsymbol{\lambda}$	: Latent Heat of Vaporization

USING NUCLEAR ENERGY IN SPACECRAFT FOR PROPULSION AND POWER IN A MICROGRAVITY ENVIRONMENT

SUMMARY

The two of the largest developments in the 20^{th} century have been the invention of space crafts, as well as the invention of nuclear power as a source of long lasting energy. Now, both technological fields have grown immensely to structure the technology of our world. These two fields are intertwined, as the future of space exploration depends on the availability of nuclear power.

One of the largesr requirements of a space flight mission is to be able to fly to furthest reaches of the universe. In fact, flying anywhere in our Solar System besides the moon will pose a considerable challenge as the distances that need to be travelled are great. The various available means of chemical propulsion in spacecraft do not pose the necessary high specific impulse and velocity to reach these long distances. But, with the availability of nuclear power, spacecraft will have the thrust as well as the necessary power for exploration of interstellar distances.

So far, the largest achievements of the various nuclear space programs all over the world have been to use nuclear fission as a source of propulsion for rockets. Atmospheric flight as well as outer space flight has been attempted by both the Americans and the Russians. In addition, radioisotope thermal heat generation is widely used in deep space satellite such as the Cassini spacecraft to provide the necessary electrical power in the deep reaches of space.

There have been some leaps in the usage of nuclear reactors for providing thrust to the spacecraft. There are some programs underway by NASA, that plan to use nuclear propulsion for a proposed Mars Mission in 2020. Also some further plans by NASA, suggest using nuclear powered shuttles in the outer atmosphere for payload transit. In addition, some advanced concepts of using fusion as a source of power for a spacecraft have been suggested.

UZAY ARAÇLARINDA NÜKLEER ENERJİNİN TAHRİK SISTEMLERİ VE GÜÇ ÜRETİMİ İÇİN MİKROGRAVİTE ORTAMINDA KULLANIMI

ÖZET

20. yüzyıldaki iki önemli gelişme uzay araçlarının icadıyla nükleer gücün bitmeyen bir enerji kaynağı olarak bulunmasıdır. Bugün, her iki teknolojik alan da inanılmaz derecede büyüyerek dünyamızın teknolojik gelişimini şekillendirmişlerdir. Bu her iki alan birbiriyle ilişkilidir, zira uzay araştırmalarının geleceği nükleer gücün var olmasına bağlıdır.

Bir uzay uçuşundan beklenen en önemli özellik, evrenin en uzak köşelerine ulaşabilmektir. Seyahat edilmesi gereken mesafeler gerçekten büyük olduğundan esasında, güneş Sisteminde Ay dışında herhangi bir yere seyahat etmek ciddi bir problem yaratır. Uzay araçlarında kullanılan kimyasal tahrik yöntemleri, bu uzak mesafeler için gerekli spesifik impuls ve hızları sağlayamamaktadır. Fakat nükleer gücün varlığı ve kullanımı ile uzay araçları için gerekli itki ve yıldızlar arası mesafeler için gerekli araştırmalar için gerekli enerji sağlanabilir.

Dünyadaki çeşitli nükleer uzay programlarının elde ettiği en büyük başarı nükleer fisyonun roketler için bir itki gücü olarak kullanılması olmuştur. Hem atmosferik uçuş hem de dış uzay uçuşları Amerikalılar ve Ruslar tarafından denenmiştir. Buna ek olarak, radyoizotopla ısıl enerji üretimi, Cassini Uzay Aracı gibi derin uzay araştırmaları yapabilen uzak mesafe uzay araçlarındaki sistemler için elektrik üretiminde kullanılmaktadır.

Nükleer reaktörlerin uzay araçlarının tahrik sistemlerinde kullanımı konusunda bazı atılımlar olmuştur. NASA tarafından yürütülen ve nükleer tahrik sistemlerini “Mars 2020” görevi için kullanmayı öngören programlar mevcuttur. Ayrıca NASA tarafında nükleer güçle çalışan mekiklerin atmosfer dışı yük taşımak için kullanılması düşünülmektedir. Ek olarak füzyonun bir güç kaynağı olarak uzay araçlarında kullanımı da düşünülmektedir. Bu tez bunları detaylı olarak incelemektedir.

.

1. INTRODUCTION TO USING NUCLEAR REACTORS IN SPACECRAFT

One of the most important innovations in the world is definitely the invention of nuclear reactor. In essence, a nuclear reactor allows the usage of a nuclear fission reaction, so that heat energy can be generated, which is then channeled toward a steam turbine where in turn is used to generate electricity. Nuclear reactors have been used widely since the 1960's to generate electricity all over the world ranging from the United States to USSR (Russian Federation). In addition, nuclear reactors have been used to generate electricity and propulsion in navy vehicles.

In fact, it is very common to use nuclear reactors in the navy ships of the United States, Russian Federation, France, and the United Kingdom. It is also common to use nuclear propulsion for power in aircraft carriers, submarines and in battleships along with missile cruisers. Hence, as it can be seen, nuclear reactors have been widely used both in civilian applications, as well as in military applications all over the world since the 1960's.

One of the most promising applications for nuclear technology is its use for space vehicles and space applications. This is a promising concept, as it is possible to use nuclear power and nuclear reactors in variety of ways in space vehicles and with other miscellaneous space applications. Nuclear energy is developing in several fronts, as it can be used at the cutting edge of science and technology; so that it allows the usage of advanced technology to make spacecrafts become more usable in many aspects. Nuclear technology is becoming widely employed in spacecraft applications. It is expected that in time, nuclear propulsion will dominate all of the space programs in the world.

1.1 Purpose of the Thesis

In this thesis, the purpose is to highlight the major aspects of nuclear propulsion in spacecraft. The main objective of this study is to show that nuclear propulsion can become the main solution to traveling long distances. The thesis will attempt to show that by using nuclear reactors for propulsion in microgravity environment, it will be possible to travel to Mars or Jupiter with ease. In addition, even interstellar travel will be possible with advanced methods of nuclear propulsion in deep space.

1.2 Background

The main operating parameters of the spacecraft can be influenced by the use of nuclear reactors in spacecraft. It is possible to use a nuclear reactor for two different applications in a spacecraft:

1) Nuclear energy can be used for power generation for its onboard navigational systems, life support systems, as well as for logistics systems by producing electricity through nuclear means. It is possible to produce this electricity through thermionic generation or by using a reactor that allows for the production of electricity through various thermodynamic schemes.

2) Nuclear energy can be used for the generation of forward thrust by the Newton's Second Law. It is important to note that the heating of a propellant gas such as hydrogen will be accomplished with efficiency through the use of a nuclear reactor aboard the spacecraft. Then this heated propellant will be allowed to exit the spacecraft through a converging diverging nozzle so that the reaction of the momentum transfer will create a steady thrust for the spacecraft. Then the spacecraft will continue to move unimpeded by the use of the Newton's First Law of Mechanics.

1.3 Objective of the Thesis

This thesis is aimed to implement the fact that using nuclear power in spacecraft can be imperative for many long range missions that are still being designed. Thus, the reasons for using nuclear propulsion instead of chemical propulsion will be laid out in detail. In addition, the primary aspects of nuclear propulsion such as heat transfer, fluid dynamics, neutronics, shielding and materials will be mentioned.

2. METHODS OF USING NUCLEAR ENERGY IN SPACECRAFT

2.1 How to Use Nuclear Energy in Spacecraft?

There are three main modes for utilizing nuclear reactors in spacecraft. These can be stated as:

a) The Usage of Nuclear Batteries for Supplying Thermionic Power to the Spacecraft

b) The Usage of Nuclear Reactors for Generating Electricity to the Spacecraft

c) The Usage of Nuclear Reactors for Generating Propulsion for the Spacecraft.

In this thesis, mainly the usage of nuclear reactors for generating propulsion for spacecraft will be explored. In addition, specific case and design examples will be provided in detail.

2.2 The Usage of Nuclear Reactors for Generating Electricity in Spacecraft

It is possible to use nuclear reactors to generate electricity for the onboard systems in any spacecraft. This includes all the onboard systems of the spacecraft such as navigation, cooling, environmental conditioning, onboard computers, onboard labs, onboard storage systems, onboard galley and other systems that will require uninterrupted electricity 24 hours a day. Any interruption in the electrical power systems of these systems can have catastrophic consequences for the spacecraft and its crew. The principles for this are simple and the nuclear reactor is used through a thermodynamic cycle to generate electricity for the spacecraft.

2.3 The Usage of Nuclear Reactors for Generating Thrust in Spacecraft

A more exotic way of using nuclear reactors in spacecraft is to use nuclear power for propulsion. A space vehicle will need something to propel it in a microgravity environment in order to move. The Newton laws of equations govern this process, as a reaction needs to take place in order to propel the spacecraft. Once the spacecraft has gained momentum in a microgravity environment, the forward momentum will continue even when no further force is applied. This way as seen in Figure 2.1, the heated gas by nuclear reactor that is being propelled, will create the necessary momentum for the spacecraft.

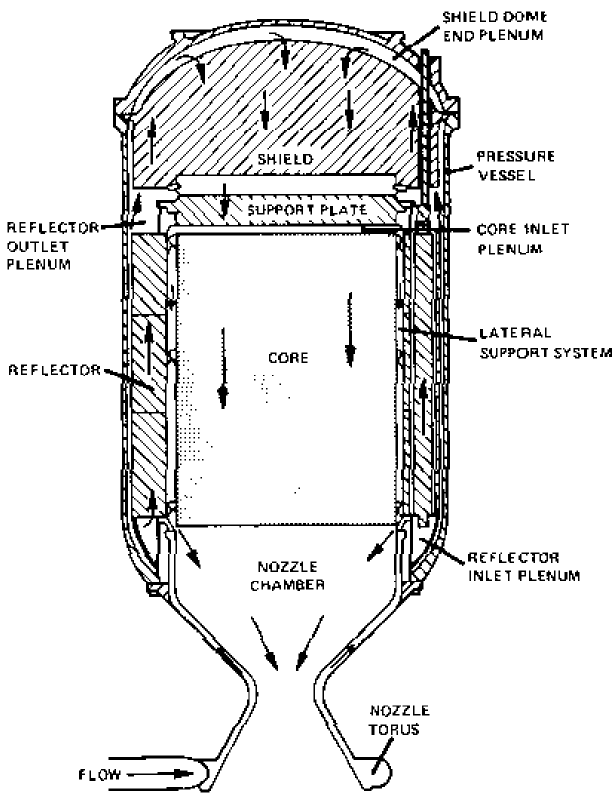


Figure 2.1 : Nuclear Reactor as a Heat Source in Rockets (Kulcinski, 1996)

2.4 The Usage of Nuclear Reactors for Take Off in Rockets

Although, this can be considered as part of the section above (generating thrust and propulsion), it is also important to state that using nuclear power for take off requires more different techniques as compared to just simple propulsion in space. This is the hardest application associated with using nuclear reactors for spacecrafts.

In this application, the nuclear reactors are employed to generate a series of small explosions that will be used to propel plasma from the rocket exhausts in order to propel the spacecraft upward. This can also be deemed as a plasma drive and is usually found to be suitable for rockets rather than space vehicles themselves. However, this method will not be discussed in this thesis, since the use of nuclear energy for atmospheric flight has been found to be too risky and banned by the United Nations.

2.5 The Advantages of Using Nuclear Technology in Spacecraft

Using nuclear reactors in spacecraft as detailed in Figure 2.2, can have many benefits to the operation of spacecraft in a microgravity environment (such as the weightless environment in space). Any of the three methods detailed above will provide different aspects of spaceflight to the designer and the advantages can range from in a wide spectrum from technical to financial.

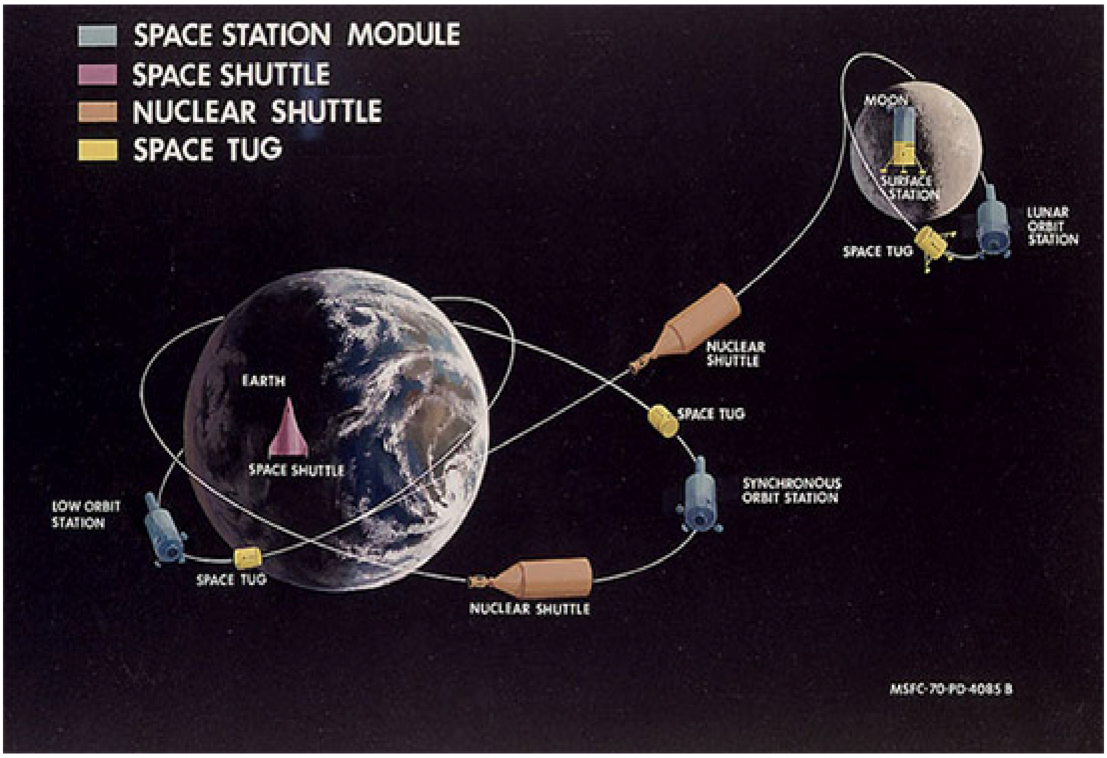


Figure 2.2 : Usage of a Nuclear Shuttle (adapted fromURL-1)

Some advantages of using nuclear power in spacecraft include:

- The ability to mount long range and long term expeditions to outer space becomes more plausible with nuclear power. Many of the probable NASA missions were not seen to be feasible due to the fact that the amount of power required was immense. Furthermore, the standard power production techniques in a microgravity environment were limited in scope.

- The dependency on solar power in space missions would be greatly reduced, as almost all of the spacecraft's avionic and navigational systems can be made to work solely with the power produced from the nuclear reactor.
- Nuclear reactor systems are more durable than standard power production systems in spacecraft.
- Nuclear propulsion is used to create speeds that are much more higher than what is normally possible with chemical propulsion techniques. Therefore, this allows for mankind to travel to further distances using this technology. The transit times in the solar system alone can be greatly reduced into usable time frames. It will be possible to launch space missions that last 25 to 50 years in duration.
- Without changing fuel, the nuclear reactor within the spacecraft can be used for decades without having fuel constraints.
- The heating problem of the spacecraft in space as well as the oxygen conversion process is taken care of. Besides the electricity generated by the nuclear reactor, the heat from the reactor core can be utilized to generate continuous heat for the spacecraft.
- Propelling the spacecraft with high velocities is possible by heating a hydrogen-slush-fuel mix that can be directly vented to the outside, from the nozzle of the spacecraft. As compared to standard thruster methods, this can help conserve fuel and attain higher speeds.

3. THE HISTORY OF SPACE PROPULSION IN THE WORLD

The idea of using nuclear power and nuclear reactors in spacecraft and in space applications is not a new idea. The application of nuclear technology in spacecraft has been experimented since the 1960's. Both the United States and the USSR Governments utilized nuclear reactors in various spacecraft designs.

Some parts of the research were devoted to using nuclear propulsion and nuclear thrust techniques in rockets for extra atmospheric flight. Other parts of research were dedicated to using nuclear power generation in satellites for electricity. Although in theoretical aspects, the Russians were able to do more interesting things with these concepts; however in practical aspects, the Americans were able to work on miniaturized nuclear reactors suited for spacecraft in Los Alamos National Laboratory in Pasadena, Texas.

3.1 The Usage of Nuclear Reactors in the American Space Program

NASA started utilizing these technologies for both nuclear propulsion techniques and also for generating electricity to power up the systems in spacecraft to help overcome the power limitation problems in spacecraft and in satellites. One of the first crude nuclear programs that were implemented was the usage of radioisotope thermoelectric generators (RTG) in spacecraft. Radioisotope thermoelectric generators (RTG) utilized the heat generated from the high decay of plutonium-238 (^{238}Pu). This was useful for generating electricity from the heat by using solid state thermocouples. These solid state thermocouples act as a thermoelectric element that can be used to generate electricity. With the recent technology, it is possible to generate power up to 1000 watts by using radioisotope thermoelectric generators (RTG). This limited nuclear technology has been used in over 25 NASA missions including the Apollo and the Voyager missions. In fact, the reason for success behind the Voyager mission that was destined to travel to the outer solar system was dependent on this technology (IAEA, 2005).

Advanced fissions systems in spacecraft were also utilized by NASA. An important mission that was a defining factor was the US SNAP-10A satellite that was launched in the year 1965, which was a 45 kWt thermal nuclear fission reactor which produced a total of 650 watts using a thermoelectric converter and the satellite happened to be operational for 43 days. Hence, as early as 1965, NASA was able to demonstrate that it was possible to use a nuclear fission reactor that was stable and that was able to operate in a microgravity environment without any major difficulties. Although, there were some problems associated with this approach, it was a good demonstration of the possibility of using nuclear fission reactors in spacecraft. (Duggins, 2007).

SP-100 reactor was also an important marker in the NASA Nuclear Space Program in 1985. This system employed a 2 MWt fast reactor unit and thermoelectric system which was able to deliver up to 100 kWe power to use for multi systems in spacecraft. It utilized fuel made out of uranium nitride and it was cooled by lithium. This was an important development, since cooling a fission reactor in a microgravity environment is not a small task, due to the fact that a simple water cooling system can not be employed as classical reactor systems do. Either plasma cooling or gas cooling must be employed to achieve this purpose. Hence, the SP – 100 reactor seen in Fig. 3.1 was a good demonstration for this concept (IAEA, 2005).

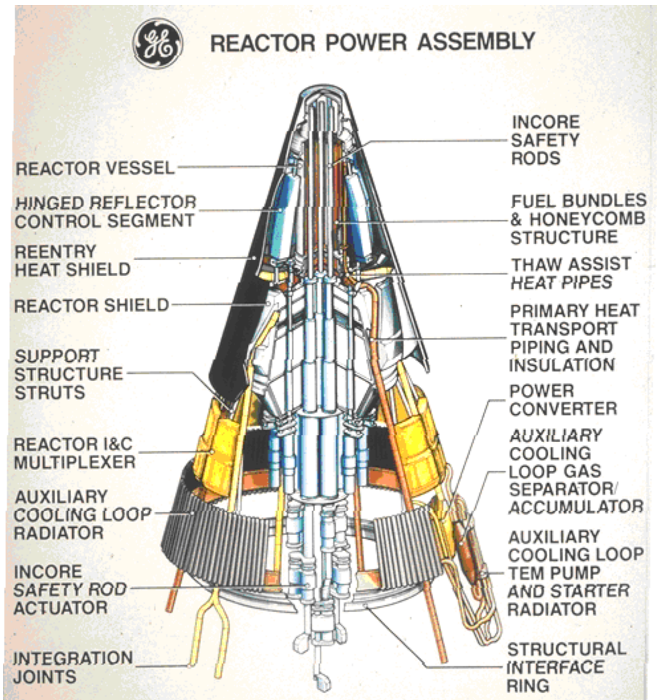


Fig 3.1: SP–100 Nuclear Reactor Assembly (Adapted from URL 1)

The NERVA rocket is also worth to be mentioned in the US space program for nuclear propulsion. NERVA stands for "Nuclear Engine for Rocket Vehicle Application". The NERVA program was initiated in 1955, by the Atomic Energy Commission of the United States and with the help of the Los Alamos National Laboratory. Then, in 1961, with the cooperation of the Westinghouse Electrical Corporation and the Aerojet General Corporation, the NERVA engine program for spacecraft was started. In essence, NERVA Reactor as shown below (Fig.3.2) is made up of a cylindrical core of uranium loaded graphite and surrounded by an insulating shell, which is a beryllium reactor and 12 rotatable drums in the reflector. The reactor and the shield are contained in a pressure vessel that is connected to a nozzle and its extension (Finseth, 1991).

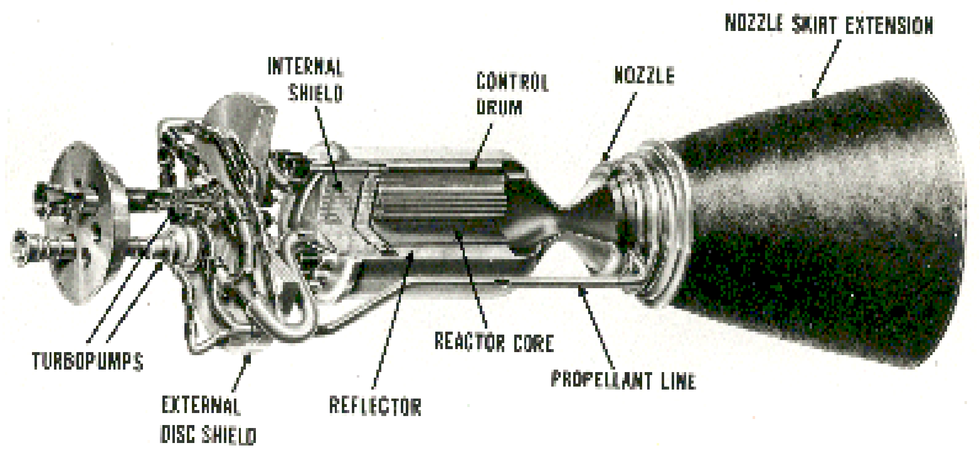


Fig 3.2 : NERVA Nuclear Engine for Rockets adapted from URL 1

As another alternative, The SAFE-400 space fission reactor was designed by NASA and by the Los Alamos National Laboratory. SAFE-400 space fission reactor or (Safe Affordable Fission Engine) is a 400 kWt HPS producing 100 kWe to power a space vehicle using two Brayton power systems—gas turbines driven directly by the hot gas from the reactor. Heat exchanger outlet temperature is around 880°C. The reactor has 127 identical heat pipe modules that are made of molybdenum, or niobium with 1% zirconium. Each has three fuel pins of 1 cm diameter, nesting together into a compact hexagonal core 25 cm across. The fuel pins are 70 cm long (fuelled length 56 cm), the total heat pipe length is 145 cm, extending 75 cm above the core, where they are coupled with the heat exchangers. The core with reflector has a 51 cm in diameter. The mass of the core is about 512 kg and each heat exchanger is 72 kg. SAFE has also been tested with an electric ion drive by NASA in the 1960's. (IAEA, 2005)

3.2 The Usage of Nuclear Reactors in the Russian Space Program

In this nuclear race for space, the Russians were not standing idle, as they were also working on parallel programs to help counteract the advancements of the Americans. The former Soviet Union was able to launch 31 low – powered nuclear fission reactors that were used in their Radar Ocean Reconnaissance Satellites (RORSAT) on Cosmos missions between the years of 1967 to 1988. They employed Romashka reactors, which utilized spectrum graphite reactors with 90%-enriched uranium carbide fuel operating at high temperatures. Later on, the Russians enhanced these fission reactors by using uranium – molybdenum fuel rods in their nuclear reactors.

These designs were followed by the Topaz reactors (as seen in Figure 3.3) with thermionic conversion systems, generating about 5 kWe of electricity for on-board uses. This was a US idea developed during the 1960s in Russia. In Topaz-2, each fuel pin (96% enriched uranium dioxide (UO_2)), sheathed in an emitter, and was surrounded by a collector that form the 37 fuel elements, which penetrated the cylindrical ZrH moderator. This in turn was surrounded by a beryllium neutron reflector containing 12 rotating control drums. NaK (a sodium-potassium alloy) was used as a coolant surrounding each fuel element (IAEA, 2005).

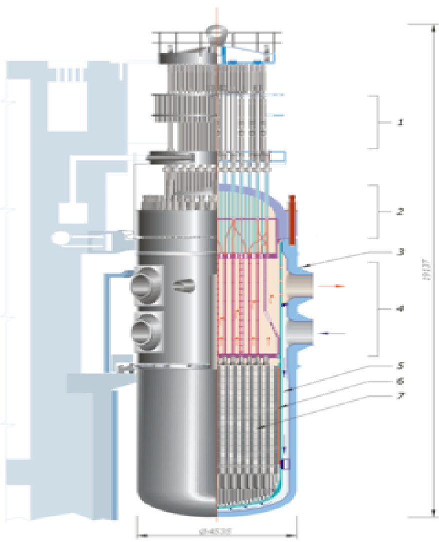


Fig 3.3 : Topaz II Nuclear Reactor for Russian Spacecraft (Kulcinski, 1996)

As compared to the Americans, the Russians explored more long range means for nuclear propulsion, but in the end, it was the American system that triumphed. However, with the end of the Cold War, the research and the practical studies on nuclear propulsion for spacecraft slowed down severely (Duggins, 2007).

As a practical result, after the 1980's, the Russian Space Exploration Program was discontinued and the Nuclear Space Program stopped as well, too. However, the Americans continued the line of research at Los Alamo National Laboratory. However, the practical applications of using nuclear rockets were severely limited after the 1990's as seen on Fig. 3.4. Nevertheless, NASA is in the process of reviving its nuclear rocket program, while the Russians have discontinued its use permanently (IAEA, 2005).

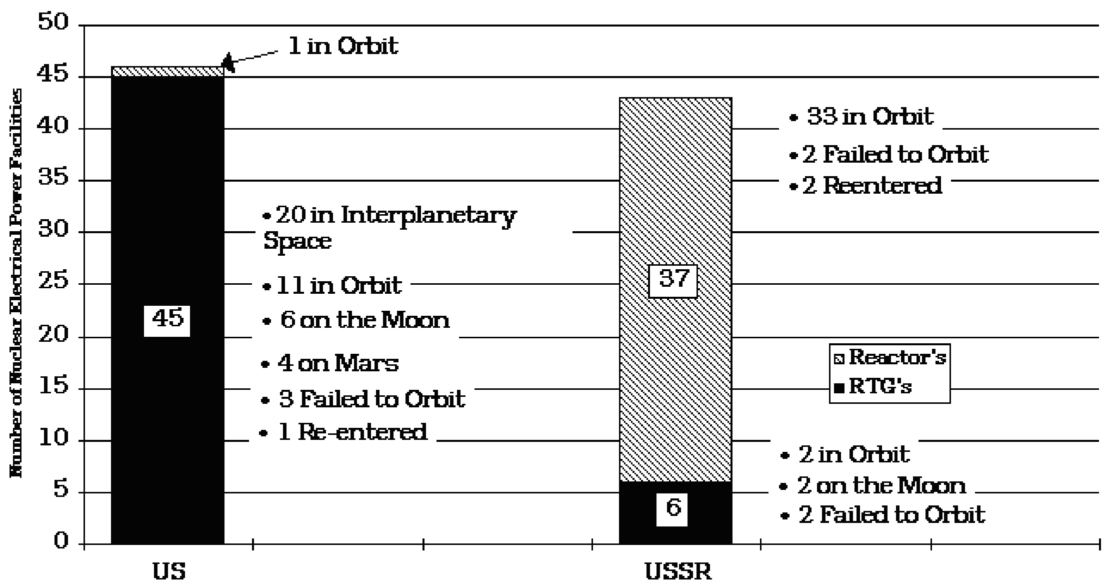


Fig 3.4: American and Russian Nuclear Space Vehicle Comparison (Kulcinski,1996)

3.3 The Usage of Nuclear Reactors in the European Space Program

In the 1980s, the Europeans were also active in using nuclear technology for spacecraft and for various space missions. For example, the French ERATO space program considered using three 20 kWe turboelectric power systems for a space mission. This system used a Brayton cycle converter with a helium-xenon mix as its working fluid. The first nuclear space system of Europe was a sodium-cooled fast reactor that utilized uranium dioxide-fuel, which operated at 670°C, the second a high-temperature gas-cooled reactor (thermal or epithermal neutron spectrum) that worked at 840°C, and the third a lithium-cooled uranium nitride-fuelled fast reactor which worked at a temperature of 1150°C (IAEA, 2005).

The contribution of the European Space Programs to the totality of Nuclear Propulsion and Nuclear Power Programs for space was severely limited. In fact, the main momentums for the nuclear space efforts were the Americans and the Russians with their separate space programs. It is possible to see the chronology of the nuclear propulsion development on the chart below at Figure 3.5.

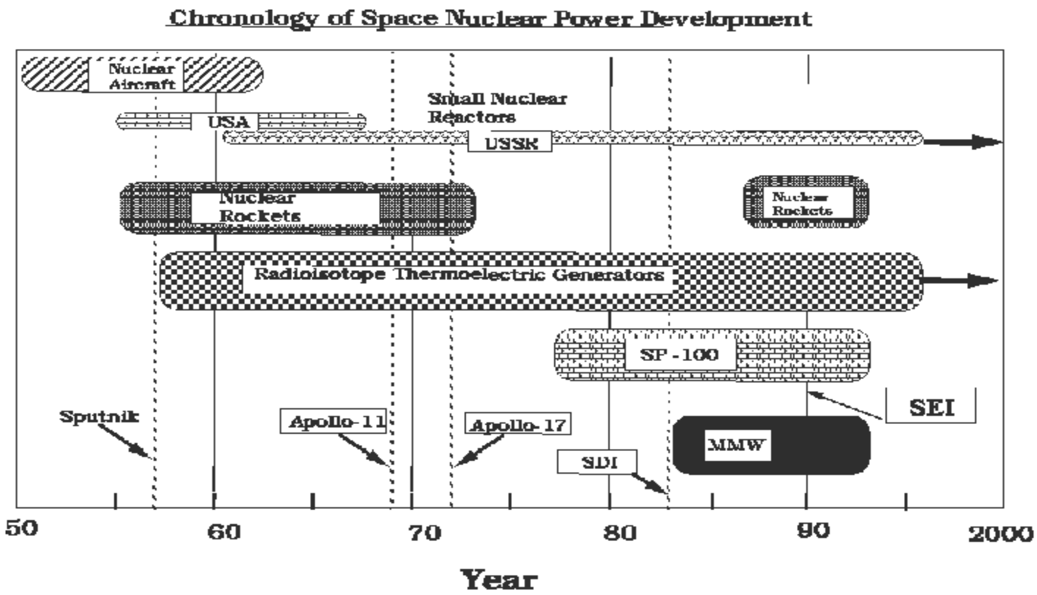


Fig 3.5 : Chronology of Space Power Nuclear Development (Kulcinski, 1996)

The technical comparison of the different space reactor power systems can be seen in Table 3.1.

Table 3.1 : Comparison of Different Space Reactor Systems (Kulcinski, 1996)

Space Reactor Power Systems

	SNAP-10	SP-100	Romashka	Bouk	Topaz-1	Topaz-2	SAFE-400
	US	US	Russia	Russia	Russia	Russia-US	US
dates	1965	1992	1967	1977	1987	1992	2007?
kWt	45.5	2000	40	<100	150	135	400
kWe	0.65	100	0.8	<5	5-10	6	100
converter	t'electric	t'electric	t'electric	t'electric	t'ionic	t'ionic	t'electric
fuel	U-ZrH$_x$	UN	UC_2	U-Mo	UO_2	UO_2	UN
reactor mass, kg	435	5422	455	<390	320	1061	512
neutron spectrum	thermal	fast	fast	fast	thermal	thermal/ epithermal	fast
control	Be	Be	Be	Be	Be	Be	Be
coolant	NaK	Li	none	NaK	NaK	NaK	Na
core temp. °C, max	585	1377	1900	?	1600	1900?	1020

4. BASIC AEROSPACE EQUATIONS FOR ROCKETS AND SPACECRAFT

4.1 The Essentials of Propulsion in Spacecraft

For any spacecraft to function, these factors have to be considered for propulsion:

- ✓ The distance to the destination expected to travel with the spacecraft.
- ✓ The time of the trip that is projected for the spacecraft
- ✓ The amount of propellant fuel that can be taken for the trip
- ✓ The total volume and the mass of the spacecraft
- ✓ Mass of the payload that will be transferred with the spacecraft
- ✓ Gravitational force fields acting upon the spacecraft
- ✓ The thrust mode as well as the acceleration mode of the spacecraft
- ✓ The decision as to whether the spacecraft will carry live crew or just payload for the duration of the trip
- ✓ The decision on whether the spacecraft will return on the trip back or whether it will be abandoned (such as in interplanetary missions)

In order to make these determinations, the functional parameters of the spacecraft will need to be evaluated and calculated. For this, it is essential to understand how a rocket works and this is illustrated in Figure 4.1. The figure shows how the propellant is burned and how the mass of the rocket changes as a result. The mass proportion of the rocket changes as the propellant is exhausted.

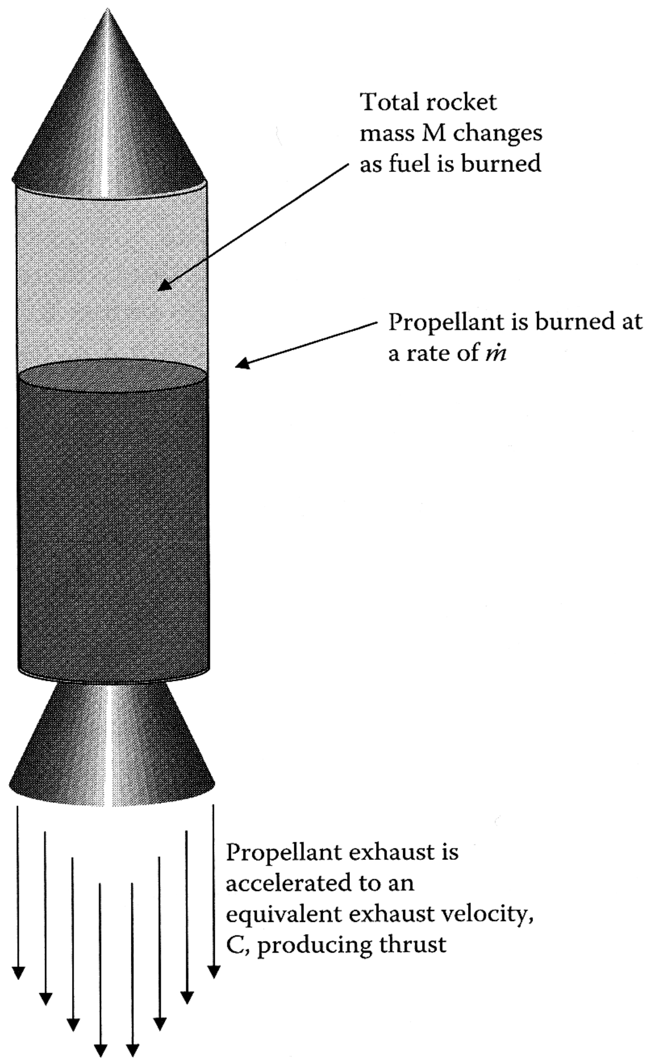


Fig 4.1 : Simple Illustration of How a Rocket Works

4.2 Specific Impulse as the Main Parameter for the Spacecraft

Specific Impulse, I_{sp}, is the most important parameter in a spacecraft, as it depicts the performance characteristic of the spacecraft. Specific impulse can be defined as ratio of the thrust to the weight consumption rate of propellant. Hence, higher specific impulse will cause the spacecraft to have more propulsion capability, as well as more speed capability as seen in Figure 4.2. High specific impulse will allow for using less propellant. Thus, the main purpose of the aerospace engineer is to always design a spacecraft with the highest specific impulse possible. In this thesis, *the Specific Impulse* is used as the main parameter by which the performance of the nuclear powered spacecraft can be measured and categorized.

The equation for the specific impulse is:

$$I_{sp} = \frac{V_e}{g} \quad \textbf{(4.1)}$$

The parameters are defined as:

V_e = The exhaust velocity of the spacecraft (m/s)

g = Gravity Acceleration constant (9.81 m/s^2)

Hence, for example, for a spacecraft engine with a specific impulse of 500 seconds, there will be an exhaust velocity of 4905 m/s. Alternatively, the space shuttle has a specific impulse of around 363 s at sea level. (Taylor, 2009)

Although it is not used as widely as specific impulse, the total impulse of the spacecraft can be defined as:

$$I_{tot} = F \cdot t \quad \textbf{(4.2)}$$

F = Thrust of the Spacecraft (Newton)

t = The amount of time the spacecraft accelerates with the thrust (seconds)

Thrust is defined as a function of the propellant mass flow rate $\dot{m}$ and the exhaust velocity V_e:

$$F = \dot{m} \cdot V_e \quad \textbf{(4.3)}$$

If the space shuttle is used as an example, the parameters of its operation can be found by using Equations 4.1, 4.2 and 4.3. For the Space Shuttle has a specific impulse of 363 seconds at sea level. It is possible to calculate the Space Shuttle's effective exhaust velocity as well as its mass propellant rate as follows:

V_e = 3557.4 m/sec

F = 1.8 x 10^6 N

m = 505.99 kg/sec

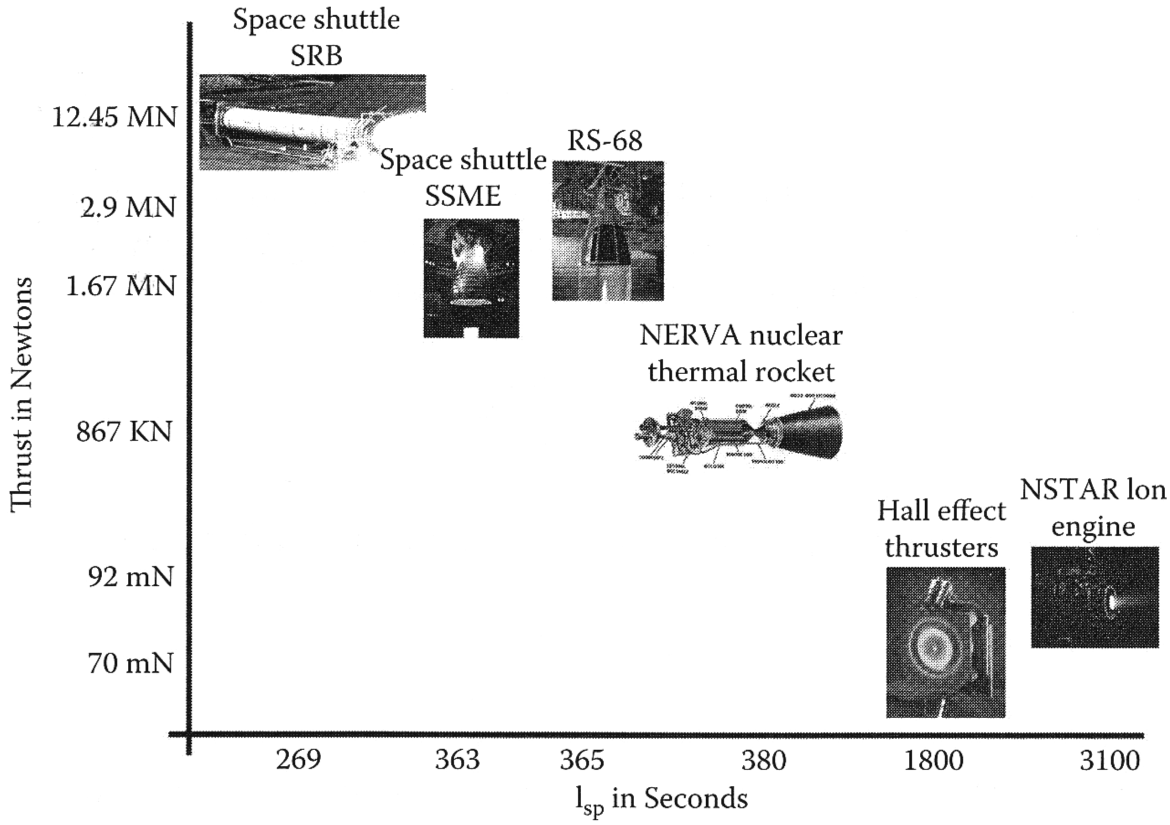


Fig 4.2 : Various Rockets with Specific Impulse per Thrust Force (Taylor, 2009)

4.3 Delta V and Mass Ratio Dependence of Spacecraft

Another important parameter is the Delta V, which can be instrumental in understanding the performance of the spacecraft's propulsion. In order to find delta V, it is essential to consider the Newton's Second Law of Momentum:

$$F = ma = m\frac{dv}{dt} \quad \textbf{(4.4)}$$

If the Equation 4.3 can be substituted for F in Equation 4.4, then, it will yield:

$$m\frac{dv}{dt} = -\frac{dm}{dt}V_e \quad \textbf{(4.5)}$$

By solving the differential equation above, the delta V equation can be found, which is known as the famous *Tsiolkovsky Rocket Equation*:

$$\Delta V = V_{exhaust} \ln \frac{M_{initial}}{M_{final}} \quad \textbf{(4.6)}$$

where

$M_{initial}$: The initial mass of the space vehicle

M_{final} : The final mass of the space vehicle (Initial mass plus Payload mass)

The Mass Ratio of the spacecraft is closely associated with the specific impulse by the exhaust velocity. As specific impulse goes down, so does Delta V, as seen in Fig 4.3.

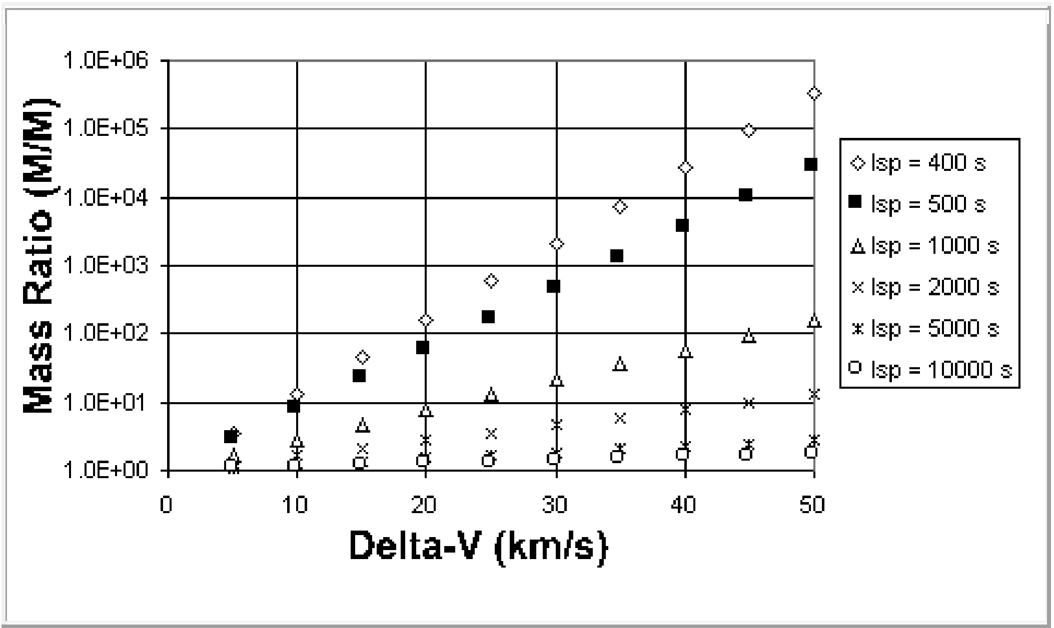


Fig 4.3 : Mass Ratio Dependence on Mission Delta-V (Taylor, 2009)

The implications of this Tsiolkovsky equation are very clear. In order to get a higher rocket speed performance (Delta V),

a) The spacecraft needs to have either a very large V_e (exhaust gas velocity) or

b) The spacecraft will need to have a very high proportion of ***m/mo***

The rocket mass ratio is a very important way of improving the rocket performance in a microgravity environment. However, as seen in the graph above, large changes in mass ratio have only a relatively small effect on the velocity of the rocket due to the natural logarithm operation in the corresponding term.

The payload capacity is also important, since the space vehicle would be required to carry a payload (whether it be scientific or technical instrumentation or a crew- the crew is also considered as the payload). Hence, this can make the juggling quite difficult, as seen in Table 4.1, although in essence it is possible to increase the amount of fuel inside the spacecraft to increase the proportion of ***m/mo*** (Sutton, 1992).

Table 4.1 : Mass Ratio Versus Fuel Usage in Spacecraft

m/mo	Fuel %
5	80
6	83.3
7	86
8	87.5
9	89.2
10	90
15	93
20	95
30	97

However, increasing the fuel is not always an option and thus that can be constricting in some long range missions. For example, in the case of the Viking spacecraft, from NASA Spaceflight Database, the following operating parameters for Viking spacecraft can be obtained:

Payload: % 1

Structure: % 16

Fuel: % 83

Thus, from the Table 4.1, this will give *m/mo* mass ratio of 6. With a Delta V rocket velocity of 5 km/sec, and by using Equation 4.6, there is a need for a minimum gas exhaust velocity of 2.8 km/sec to achieve the required performance.

4.4 Advantages of Using Nuclear Propulsion for Higher Specific Impulse

Therefore, it is evident that, it is advantageous to use nuclear propulsion instead of chemical propulsion in order to have a high specific impulse, as well as an agreeable mass ratio for the spacecraft. The maximum I_{sp} (specific impulse) which can be achieved with chemical engines is in the range of 400 to 500 s (Sutton, 1992).Hence, by using nuclear propulsion, it can be possible to have much higher specific impulse that is needed for long range missions.

The above parameters mentioned above such as the *Specific Impulse*, *Delta V*, *Exhaust Velocity*, and *Final Mass* will be very instrumental in determining on how to use nuclear reactors in spacecraft for propulsion. On the following sections, it will be demonstrated that how the nuclear reactor's temperature and its mass can affect the above parameters and how the necessary plasma exhaust can be produced, in order to create a reverse momentum transfer in a microgravity environment.

The workings of a spacecraft and its nuclear reactor will be very different in a weightless microgravity environment as compared to terrestrial applications. However, it is important to create a correlation of the nuclear reactor equations with the aerospace thrust equations in order to create a workable mission profile that can be used for long range missions requiring high velocity and high travel distances with a constant acceleration profile. (IAEA, 2005)

In the chart below on Fig. 4.4, the transit times needed to reach various destinations in the solar system can be seen. The scale of the distances and the time involved can help to highlight the importance of using nuclear propulsion in spacecraft.

Destination	**Jet 600 miles/hour 965 Kms/hour**	**Rocket 25,000 miles/hour 40,250 Kms/hour**	**Ray of Light 186,282 miles/hour 299,792 Kms/hour**
Sun	17 years	5 months	8 minutes
Earth's Moon	15 days	9 hours	1 second
Mercury	9 years	3 months	4 minutes
Venus	5 years	1 month	2 minutes
Mars	7 years	2 months	3 minutes
Jupiter	70 years	2 years	43 minutes
Saturn	141 years	3 years	1 hour
Uranus	305 years	7 years	2 hours
Neptune	509 years	12 years	4 hours
Pluto*	506 years	12 years	4 hours
Proxima Centauri (closest star to Earth other than the sun)			4 years
Center of the **Milky Way Galaxy**			27,000 years

Fig 4.4 : Distances and Travel Times to Solar System Objects (IAEA, 2005)

4.5 Thermodynamics of the Exhaust Gases in the Spacecraft

The thermodynamics of exhaust gases play a very important role in the performance characteristics of the spacecraft. Obviously, from the tables and from the equations presented above, it is technologically and scientifically more feasible to effect the velocity of the exhaust gases as a means of increasing the speed of the rocket.

If it is assumed that the rocket engine is %100 efficient and that all of the heat liberated will be converted into kinetic energy, then an equation can be postulated. From Thermodynamic relations it is known that (Bussard, 1965):

$$U_e = \sqrt{2JQ} \tag{4.7}$$

The variables are defined as:

U_e = exhaust velocity of the gas

J = mechanical equivalent of heat

Q = amount of heat liberated in kJ / g of reaction product

Due to combustion chamber limitations and also due to expansion process limitations) using chemical means of combustion won't be able to provide the results that are needed.

It is possible to derive a better equation for the isentropic expansion of gases in order to form a better equation for the exhaust velocity of the gas.

The ideal gas equation of state is defined as (Bussard, 1965):

$$pv = \frac{RJ}{M}T \tag{4.8}$$

Where,

R= Universal gas Constant

M = Molecular weight of the gas

In this equation, it is assumed in the spacecraft that there is zero velocity in the combustion chamber and that there is an ideal isentropic expansion of the gases from the exhaust chamber.

From the equation 4.7 with the above ideal gas conditions:

$$U_e^2 = 2J(H_c - H) \tag{4.9}$$

In the above equation, the Q in the Equation 4.7 has been replaced by enthalpy H.

H = enthalpy of 1 kg of gas

Q = energy of the fuel in 1 kg of gas products

It is important to note the fact that at constant pressure Pc:

$$Q_c = \Delta H_c = C_p \Delta T_c \qquad \textbf{(4.10)}$$

Hence, by using these equations with the ideal gas equation, the stream velocity of the exhaust gas is found to be (Bussard, 1965):

$$U^2 = \frac{2\gamma}{\gamma - 1} p_c v_c \left[1 - \left(\frac{p}{p_c} \right)^{1-\frac{1}{\gamma}} \right] \qquad \textbf{(4.11)}$$

Where p_c is pressure of the main chamber, p is the gas pressure at that instant, V_c is the volume of the main chamber and γ is the latent heat of vaporization

Through the values presented above, it is possible to examine the stream velocity of the spacecraft and thus the performance of the spacecraft. This provides the means to alternate the design of the spacecraft where necessary (Sutton, 1992).

If the velocity of the exhaust gas equation is written in the best possible way so that the final velocity of the spacecraft can be seen, then if the equations 4.8 and 4.11 are combined to give:

$$U_e = \sqrt{2J \frac{\gamma}{\gamma - 1} \frac{R}{M} T_c \left[1 - \left(\frac{P_e}{P_c} \right)^{1-\frac{1}{\gamma}} \right]} \qquad \textbf{(4.12)}$$

The variables are defined as:

U_e = speed of the gas at the exhaust point

P_e = gas pressure at the exhaust

T_c = Temperature of the combustion chamber

From the Tsiolkovsky's equation, the maximum velocity of the spacecraft as the function of the velocity of the exhaust gas can be defined as (Taylor, 2009):

$$V_{\max} = U_e \ln\frac{m}{m_o} \tag{4.13}$$

Thus, by combining equations 4.12 and 4.13, it is possible to get an equation for the maximum velocity of the spacecraft:

$$V_{\max} = \sqrt{2J\frac{\gamma}{\gamma-1}\frac{R}{M}T_c\left[1-\left(\frac{P_e}{P_c}\right)^{1-\frac{1}{\gamma}}\right]}\ln m/m_o \tag{4.14}$$

Equation 4.14 gives the velocity of the spacecraft as a function of temperature. As T_c is increased, the final V_{max} will also increase accordingly. This is very useful, since the temperature is the main operational parameter of spacecraft propulsion. This way, direct link up between obtaining high temperature for faster speeds is seen with the equation above. This will become handy for understanding the role of nuclear heating of a propellant in order to get a higher exhaust speed

4.6 The Analysis of Nozzle Flow in Spacecrafts with Isentropic Expansion

In order to understand the performance parameters in spacecraft, it is essential to look at the characteristics of a nozzle flow in the spacecraft. The effective propellant exhaust velocity is determined by the thermodynamic properties of the propellant gas, as well as the geometry of the rocket nozzle and the operational conditions.

Hence, it can be stated that the effective exhaust velocity is the parametric link between the exterior ballistics of the vehicle and the interior design parameters of combustion chamber, which provides the heat for the propellant. Thus, the effective propellant exhaust velocity is linked to the thermodynamic properties of the propellant gas, the geometry used in the rocket motor nozzle, as well as the operating conditions of the rocket motor (Sutton, 1992).

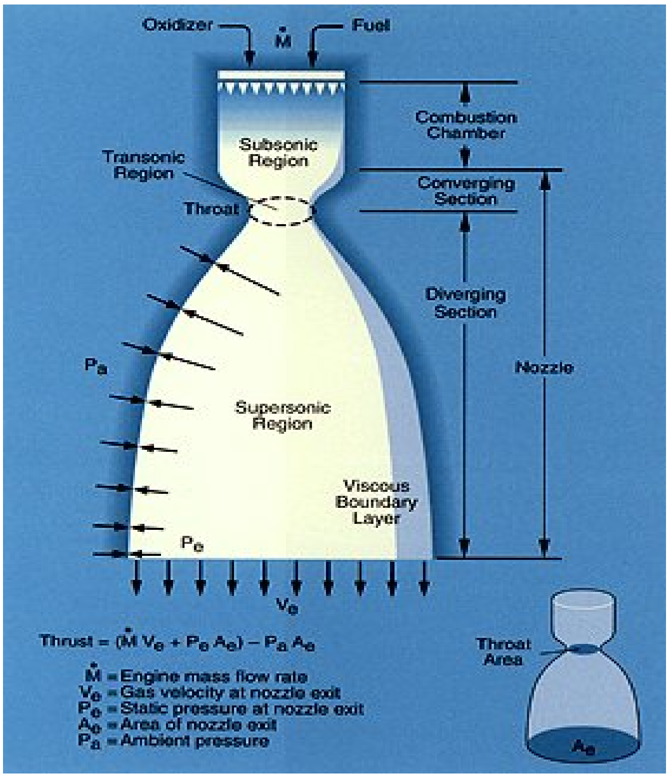


Fig 4.5 : Representation of Nozzles in an Ideal Spacecraft adapted from URL 1

In order to analyze the situation, it is considered that an isentropic expansion flow exists without any losses and with no energy input, assuming ideal gas conditions. The simplest nozzle design is the use of a converging - diverging nozzle. This is a widely used design since, convergent- divergent nozzle allows the stream velocity to reach supersonic speeds near the exit area of the nozzle. This is what makes spacecraft fly at faster speeds (Taylor, 2009).

It is possible to write the energy balance for the isentropic flow in such a nozzle with using an energy balance equation as (Bussard, 1965):

$$v_{ei}^2 - v_c^2 = 2Jg_o c_p (T_c - T_e) \qquad \textbf{(4.15)}$$

The parameters in this equation are:

J = Mechanical equivalent of heat energy

c_p = average specific heat at constant pressure over the range T_c to T_e

T_c = Combustion Chamber temperature

T_e = Nozzle exit temperature

Since, most rocket nozzles will operate with small heat transfer in relation to the kinetic energy of the propellant; it is possible to use the isentropic and adiabatic conditions to define for the expansion of gas temperature and pressure:

$$\frac{T_e}{T_c} = \left(\frac{P_e}{P_c}\right)^{\frac{\gamma - 1}{\gamma}} \tag{4.16}$$

where:

γ = ratio of specific heats at constant pressure and constant volume between e and c

P_e = Exit Pressure at the nozzle

P_c = chamber pressure

Then consequently, Carnot cycle can be used as a model of isentropic expansion for the spacecraft nozzle (Bussard, 1965):

$$\eta_c = 1 - \frac{T_e}{T_c} = 1 - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}} \tag{4.17}$$

where:

η_c = Carnot cycle efficiency

The relation between nozzle pressure and Carnot efficiency is seen in Figure 4.6:

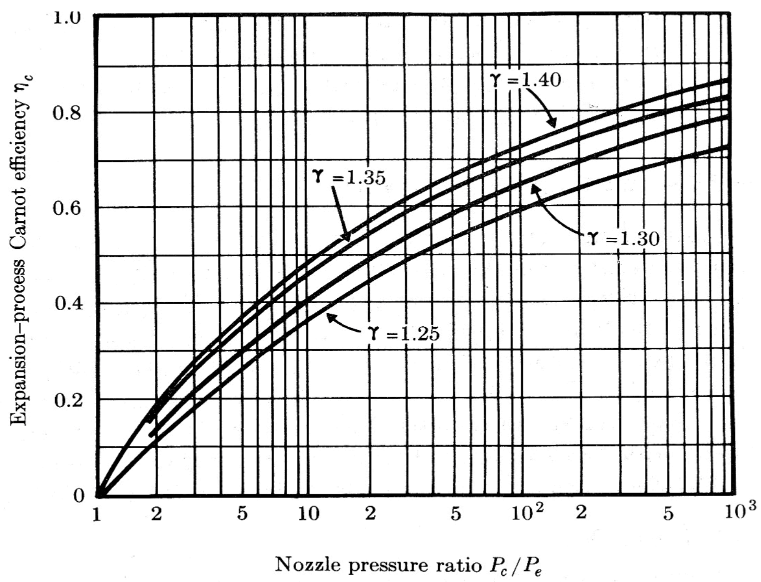


Fig 4.6 : Nozzle Expansion versus Carnot Efficiency (Bussard, 1965)

If the original energy balance Equation 4.15 is used with isentropic expansion under adiabatic conditions, then the expression will become:

$$v_{ei}^2 = v_c^2 + 2Jg_o c_p (T_c - T_e) \tag{4.18}$$

If equation 4.16 is substituted to the equation 4.18:

$$v_{ei}^2 = v_c^2 + 2Jg_o c_p (T_c - \left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}} T_c) \tag{4.19}$$

Hence, by substituting the definition of Carnot Cycle Efficiency equation of 4.17 into the above equation of 4.19, the equation can be simplified into the form:

$$V_{ei}^2 = V_c^2 + 2Jg_o c_p \eta_c T_c \tag{4.20}$$

The basic thermodynamic relations below can be used to help write the equation above in terms of change in enthalpy, per unit mass flow in propellant (Bussard, 1965):

$$v_{ei}^2 = 2Jg_o \Delta H_r \eta_e \tag{4.21}$$

ΔH_r = change in enthalpy per mass unit flow

η_e = efficiency of the energy utilization in to the nozzle

If the specific heat in the gas temperature changes are used then the following relation is defined:

$$c_p - c_v = \frac{R}{J\mathrm{M}} \tag{4.22}$$

In addition, specific enthalpy is defined as:

$$\gamma = \frac{c_p}{c_v} \tag{4.23}$$

And also by thermodynamic relations:

$$c_p = \frac{R_u}{J\mathrm{M}} \frac{\gamma}{\gamma - 1} \quad \textbf{(4.24)}$$

Hence, the ideal propellant gas exhaust velocity in the nozzle flow can be written by combining all of the equations above:

$$V_{ei}^2 - V_c^2 = \frac{2\gamma g_o}{\gamma - 1} \frac{R_u}{M} T_c \eta_c \quad \textbf{(4.25)}$$

From the above Equation 4.25, the maximum exhaust velocity can happen for a Carnot cycle efficiency of %100 with infinite expansion ratio. However, it is assumed that the speed in the rocket chamber is small as compared to exit speed at the nozzle as seen in Figure 4.7 (Sutton, 1992).

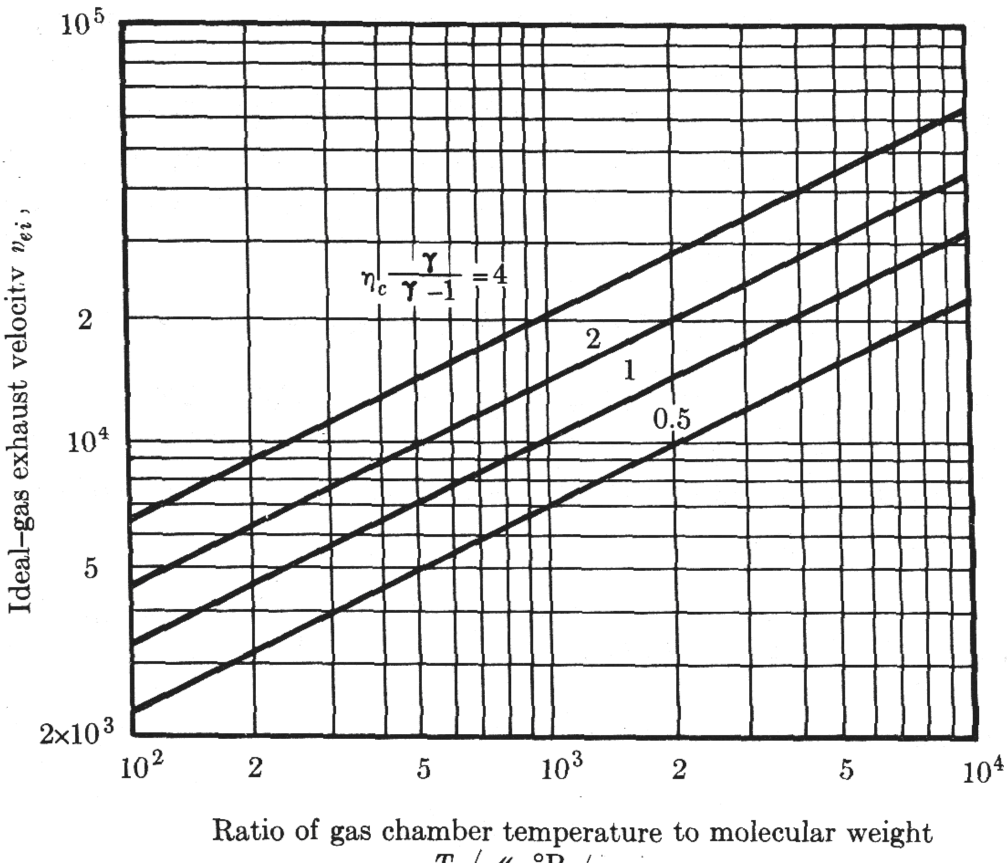


Fig 4.7 : Nozzle Exit Gas Velocity as Function of Temperature/Molecular Weight

The problem with the above equations is the fact that they have been formulated for ideal nozzles with isentropic expansion and they are modeled with Carnot efficiency.

However, in real life, the mass flow rate in a real nozzle differs from an ideal nozzle. Moreover, such other forces such as friction and turbulence can cause differences in real nozzle activity as compared with that of the spacecraft.

Through exploration of real nozzles in spacecraft, it is possible to see that the velocity equation with a velocity coefficient for real nozzles is given as (Bussard, 1965):

$$V_{ea} = V_v \left(\frac{2\gamma g_o}{\gamma - 1} \frac{R_u}{M} T_c \eta_c + V_c^2 \right)^{\frac{1}{2}} \tag{4.26}$$

One other equation for the performance characteristics of the spacecraft is the specific impulse equation for a spacecraft (or a rocket) that is operating in vacuum as related to its nozzle and propellant performance characteristics. Thus, the specific impulse for a rocket in vacuum and microgravity environment is (Bussard, 1965):

$$I_{sp} = v_v \left[\frac{2\gamma}{\gamma - 1} \frac{R_u}{\mathrm{M} g_o} T_c \eta_c + \left(\frac{v_c}{g_o} \right)^2 \right] + \frac{1}{v_d} \left(\frac{P_e}{P_c} \right)^{\frac{\gamma-1}{\gamma}} \left(\frac{\gamma - 1}{2\gamma} \frac{R_u}{M g_o} \frac{T_c}{\eta_c} \right)^{1/2} \tag{4.27}$$

The equation above will provide a cornerstone to compare nuclear propulsion with ideal propulsion in the remaining parts of the thesis.

4.7 The Analysis of Nozzle Geometry in Spacecraft

The analysis of nozzle geometry in spacecraft is very important for the understanding of the performance characteristics of spacecraft. By the proper geometrical design of the nozzle (as seen in Fig. 4.8) , the exhaust of the propellant gases will be regulated in such a way that the maximum effective spacecraft velocity can be reached. The under expansion of the nozzle as well as the over expansion of the nozzle can become problematic for the spacecraft's flight (Taylor, 2009).

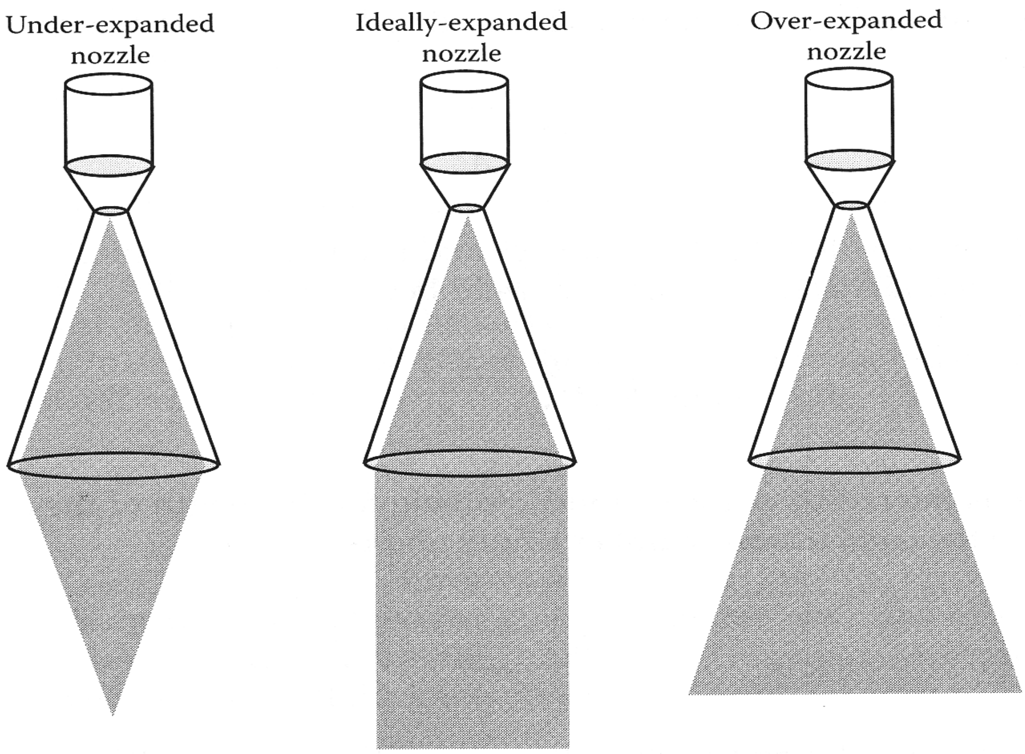


Fig 4.8 : Nozzle Expansion Geometry Effects for Thrust in Spacecraft

One of the most common types of nozzle geometry in spacecraft is definitely the convergent- divergent nozzle. By using the convergent - divergent nozzle, the propellant is heated at the combustion chamber. (In nuclear systems, the heating is done by the nuclear reactor instead of chemical combustion).

The heated propellant is converged at the nozzle throat and then it is propelled toward the divergent part of the nozzle. As the gas expands under constant temperature, it will increase its stream velocity to supersonic speeds. It is well documented by Goddard that as supersonic speeds are reached, increase in the area of the nozzle will end up and cause the flow velocity to increase as well too. Then, the flow stream will exit the nozzle in supersonic speeds, and it will give the rocket forward momentum, as a result of the reverse reaction to the propellant's momentum (Sutton, 1992).

It is the analysis and the proper design of the nozzle that can affect the final stream velocity of the spacecraft. Once the required conditions are reached through thermodynamic means, then it is only the geometry of the spacecraft that can make a difference on the exhaust stream. By changing the length and the shape of the nozzle characteristics, a proper nozzle exit profile that fits the specifications of the spacecraft can be achieved in the design as seen in Figures 4.9 and 4.10.

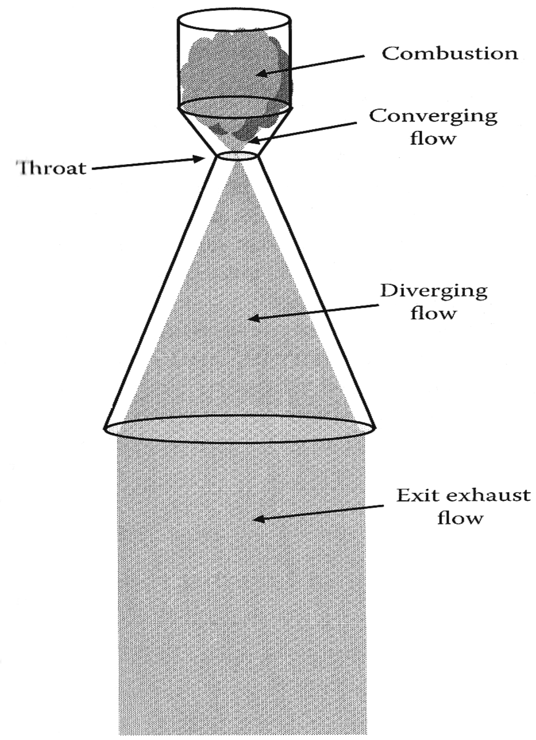


Fig 4.9 : Convergent – Divergent Flow in a Spacecraft Nozzle

The Nozzle Geometry equations will be presented in order to calculate the various lengths and areas in the nozzle. These equations will be used in this thesis to help design a sample nuclear engine and its corresponding nozzle.

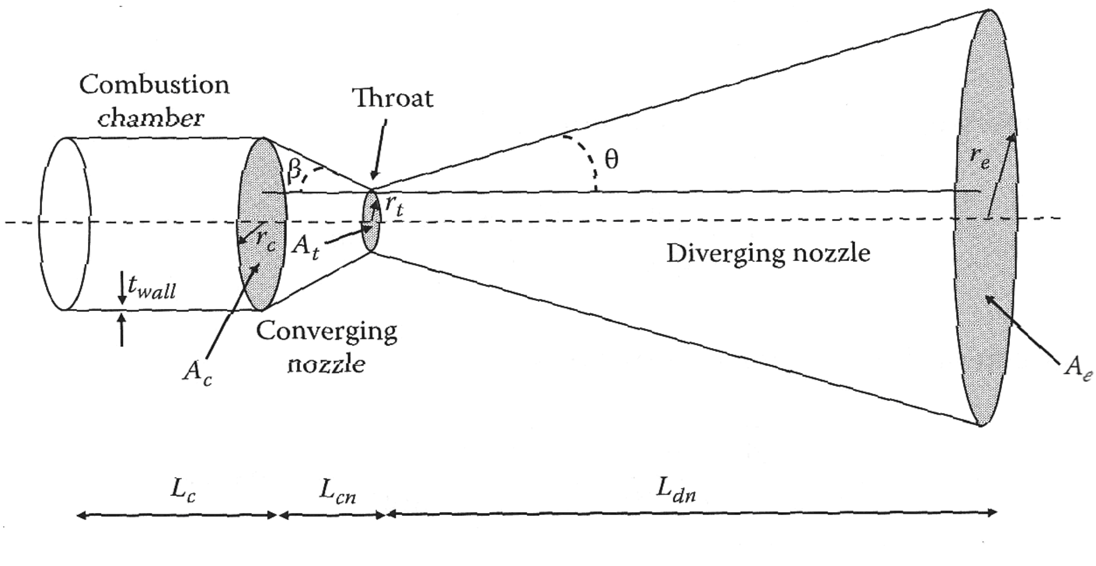


Fig 4.10 : Representation of a Spacecraft Nozzle Dimensions and Geometry

$$\dot{m} = \frac{F_{thrust}}{V_e}$$ Mass Flow in Rockets **(4.28)**

$$\varepsilon = \frac{A_e}{A_t}$$ Expansion Ratio in Nozzles **(4.29)**

$$A_t = \frac{\dot{m}}{P_c \sqrt{\gamma \left(\frac{2}{\gamma + 1} \right)^{\frac{\gamma+1}{\gamma-1}} \frac{M}{R_u T_c}}}$$ Area of the Nozzle Throat **(4.30)**

$$A_e = \pi r_e^2$$ Exit Area of the Nozzle **(4.31)**

$$A_c = 3A_t$$ Convergence area in Nozzle **(4.32)**

$$r_t = \sqrt{\frac{A_t}{\pi}}$$ Radius of the Throat **(4.33)**

$$r_c = \sqrt{\frac{A_c}{\pi}}$$ Combustion radius **(4.34)**

$$L_{dn} = \sqrt{\frac{A_e}{\pi}} \frac{1}{\tan \theta}$$ Diverging Nozzle Length **(4.35)**

$$L_{cn} = \sqrt{\frac{A_c}{\pi}} \frac{1}{\tan \beta}$$ Length of the Converging Nozzle **(4.36)**

$$L_c = \frac{A_t L^*}{\pi r_c^2}$$ Length of the Combustion Chamber **(4.37)**

It is important to note that these equations are used in every spacecraft available during the present day, ranging from the Space Shuttle to the Arianne Rockets that are used to propel satellites in to space. It is safe to assume that these nozzle characteristics will also hold true for nuclear spacecraft (Taylor, 2009)

4.7 Overview of the Performance Characteristics in Spacecraft

It is very important to stress that the most important characteristic that is used will be the specific impulse. It can be demonstrated that spacecraft, which uses a nuclear reactor to heat the propellant, can reach much more higher specific impulses as a result.

By using specific impulse, it will be possible to take a look at different nuclear reactor types and their effect on the spacecraft velocity. It can be demonstrated that using nuclear reactors for space propulsion will be the only feasible means of obtaining high distances within the solar system for exploration.

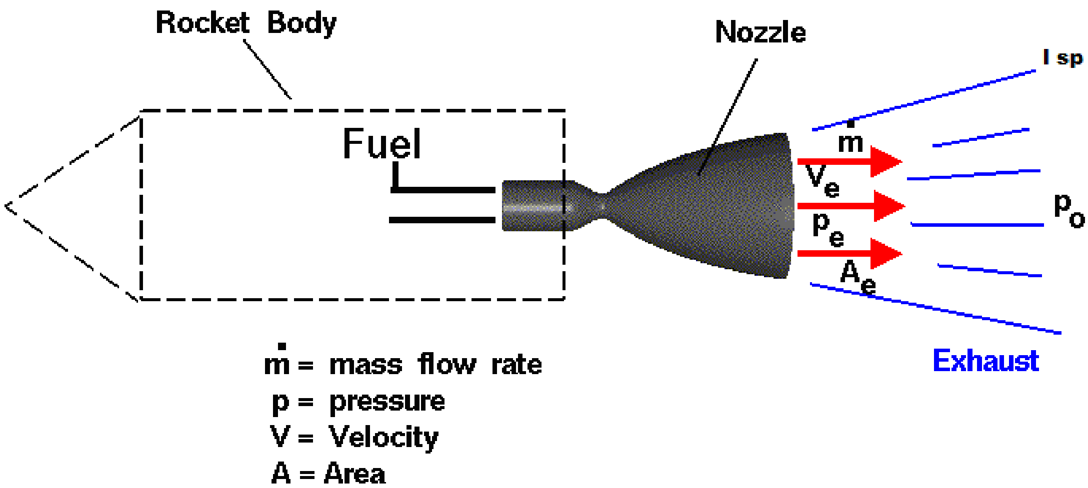


Figure 4.11 : Main Parameters of a Rocket on a Diagram

The parameters of interest are as shown in Figure 4.11, such as mass flow rate, exit pressure P_e, exit velocity V_e, nozzle throat exit area A_e, outside pressure P_o, and the specific impulse I_{sp}.

Thus, the main aerospace performance equations to use the above parameters:

- The Specific Impulse Equation 4.1, I_{sp}
- Tschiolkovsky Equation for Spacecraft Equation 4.6, Delta V
- The Thermodynamic Expression for the Velocity of the Exhaust Gas Equation 4.12,

- The specific impulse for a rocket in vacuum Equation 4.27
- Nozzle Geometry Equations 4.30 – 4.37 for the design parameters of the Converging Diverging Nozzle

5. NUCLEAR REACTORS IN SPACECRAFT FOR POWER GENERATION

5.1 The Importance of Using Nuclear Power in Spacecraft

One of the most important benefits of utilizing nuclear energy in spacecraft is the fact that it allows for large amounts of power for long durations in the spacecraft's mission profile. It would not be possible to plan any sort of a long range space mission with standard chemical rockets. Both the specific impulse of the rockets, as well as the total power output that is needed in the megawatt range cannot be sustained in any meaningful way through chemical propulsion.

However, by using nuclear reactors and other nuclear systems in spacecraft, these limitations of chemical propulsion systems are easily bypassed. For example, one of the main pillars of NASA deep space missions is definitely the Mission to Mars. NASA is interested in sending 3 to 5 astronauts aboard a spacecraft for the duration of the Mars mission (IAEA, 2005). One of the operational principles of NASA is that the mission duration should be planned to allow for the life support requirements of the human crew during the mission. In essence, it all comes down to supplying the necessary power for the spacecraft and its systems for the duration of the mission. This can be best achieved by using nuclear power for the spacecraft.

5.2 Radioisotope Thermoelectric Generators

Besides using a nuclear reactor to generate electricity for the spacecraft, nuclear power sources such as a radioisotope thermoelectric generator can be considered to supply some minimal power to the spacecraft. For deep space probes and for small exploratory satellites, this can be an efficient way to sustain the needed electricity.

The main idea depends on the fact that a nuclear decay reaction can be utilized as a heat source to create electricity through direct electric energy conversion processes. It is in essence a thermo-electric process in which the nuclear power source acts as a thermo-electron engine. This allows for supplying the necessary small to mid range power to the spacecraft (Bolonkin, 2008).

However, this technique is only useful for powering onboard computers and onboard navigational systems. It is not useful for propulsion as in the following sections of the thesis. Nevertheless, for exploratory craft such as NASA's Cassini spacecraft, the usage of radioisotope thermoelectric generators as a source of power for onboard electricity is very frequent and useful.

For example, the Cassini spacecraft seen below in Fig. 5.1, uses a 7.72 kg of Pu 238 source to create an electricity of 888 watts. This electricity is generated constantly and it powers its onboard systems (IAEA, 2005).

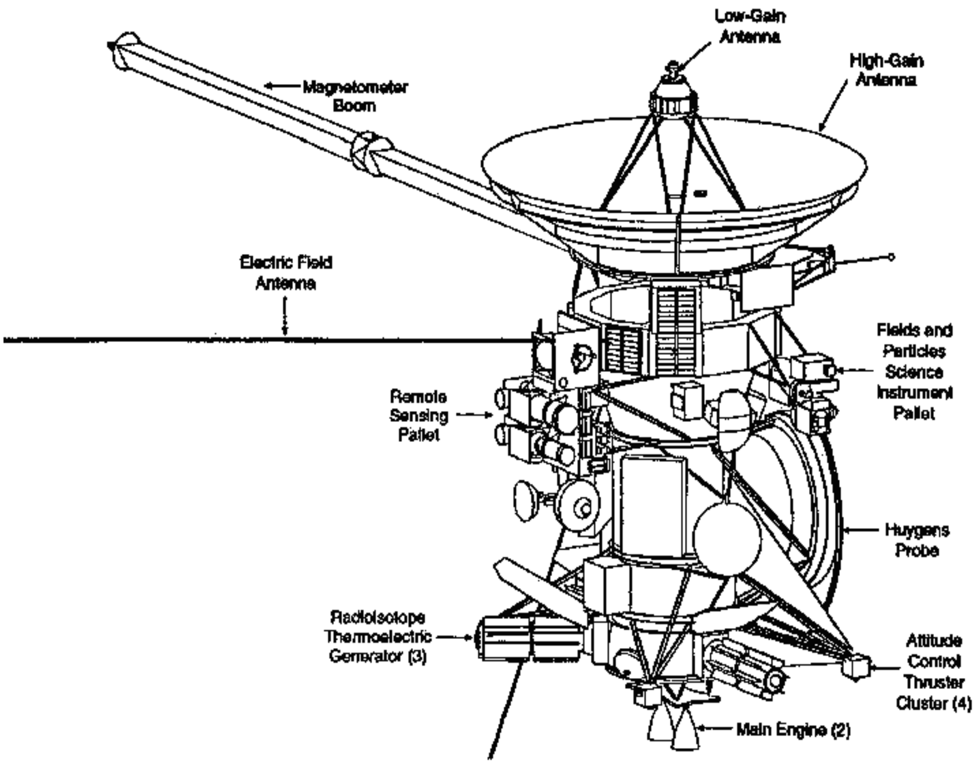


Figure 5.1: Cassini Probe with Radioisotope Thermoelectric System

The heat difference between the heat source and the cooler temperature in the opposite plate is used to create electrical power, as shown in Figure 5.2. In fact, thermoelectricity is a reliable way of converting heat energy directly into electricity, since it does not have any moving components.

In addition, by using a radioisotope source, a long term heat source for the spacecraft is guaranteed. Hence, as a result of this, many interplanetary probes such as the

Voyager deep space probe probes utilize radioisotope thermoelectric generators to constantly produce electricity for their onboard systems.

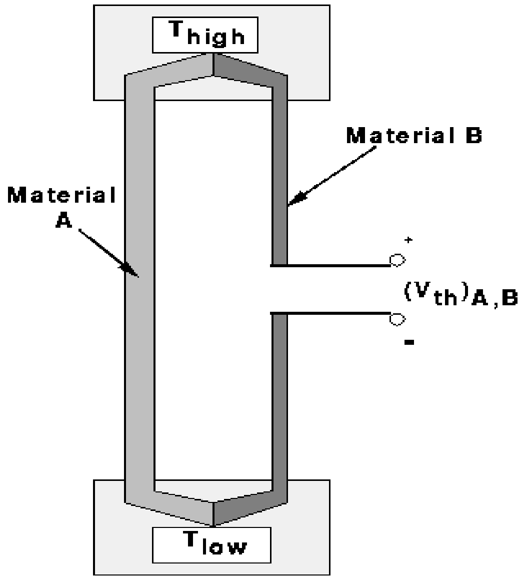


Figure 5.2 : Representation of Thermoelectric Power Generation

NASA research shows that by using this method shown in Figure 5.2, spacecraft can be powered by nuclear processes so that enough electricity can be supplied for 20 to 50 year duration missions.

5.3 Usage of Nuclear Reactors to Generate Electricity for the Spacecraft

When planning long range missions such as a trip to Mars; it will be necessary to plan for at least 2 years of duration in a mission. In a standard Mars mission, it is important to think about at least 230 days for going to Mars and another 230 days for returning from Mars, along with 10 to 20 days for detailed exploration. Hence, the mission profile would need a window of 500 days and the means to supply the necessary power for these 500 days (IAEA, 2005).

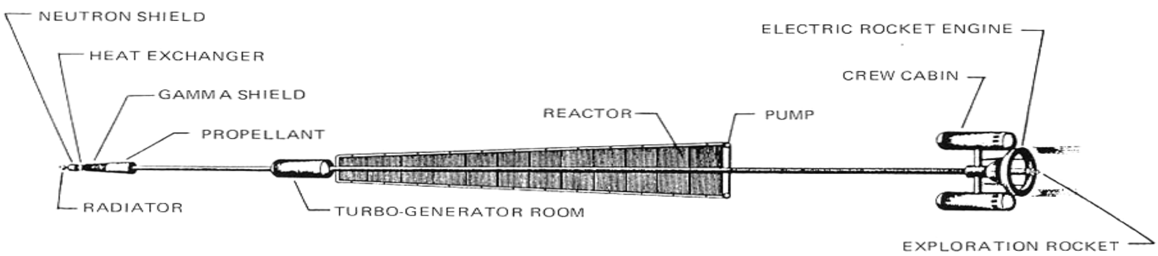


Figure 5.3 : Classic Electric Rocket Engine with a Nuclear Reactor (IAEE, 2005)

In such a long mission profile, the usage of thermoelectric systems will not be sufficient. Instead, it is essential to employ more powerful nuclear systems such as a nuclear reactor with a proper thermodynamic cycle, (like the Brayton or Rankine Cycle) to provide the necessary power output to the spacecraft. The important consideration is to design suitable reactor with appropriate power output that is in attenuation with the mission profile as seen in Figures 5.3 and 5.4. This means of generating nuclear power for electrical energy in a microgravity environment is a challenging task, since the nuclear reactor depends on gravity for control and stability.

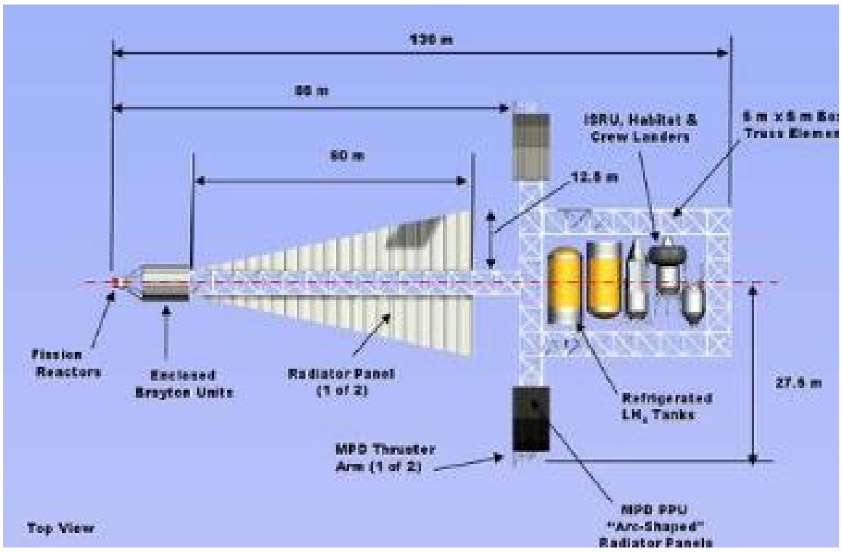


Figure 5.4 : Sample Nuclear Electric Spacecraft Design (IAEE, 2005)

5.5 The Usage of Nuclear Reactors to Generate Power for Ion Propulsion

Ion propulsion is considered as an exotic means of generating spacecraft thrust with the aid of a nuclear reactor. This type of propulsion has great capability for long range missions and thus it is on the drawing board for NASA's future interstellar missions. One of the biggest obstacles for long range missions is the storage of the propellant during the flight. Ion propulsion has been proposed as a solution to this problem, as no chemical propellant needs to be stored aboard the spacecraft for this technique.

In essence, ion propulsion generator (as seen in Fig. 5.5) creates an electrical field by generating electricity with the use of a nuclear reactor. Once this electricity has been created, it can be used to induce a magnetic field of charged particles. Then these charged particles are accelerated within the chamber and they are discharged from the rocket nozzle. NASA research shows that by utilizing this method, long interstellar ranges can be achieved and moreover very high and sustainable specific impulses can be attained.

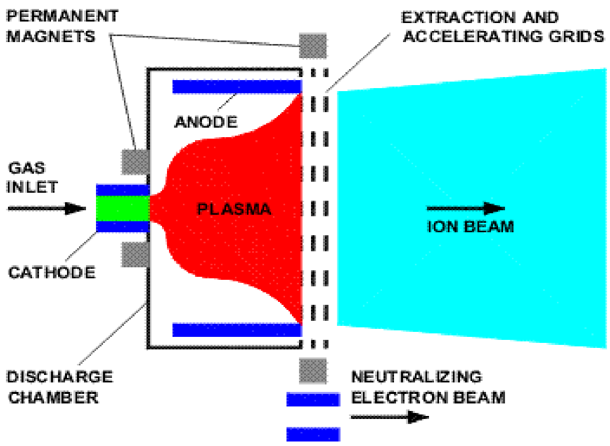


Figure 5.5 : Ion Propulsion Engine Schematic adapted from URL - 1

A nuclear electric propulsion system (NEP) uses a nuclear heat source coupled to an electric generator. The PPU (Power Processing Unit) converts the electrical power generated by the nuclear power source into the power required for each component of the ion thruster. It produces the voltages required by the ion optics and discharge chamber and the high currents required for the hollow cathodes. The PMS (Propellant Management System) controls the propellant flow from fuel tank to the thruster and hollow cathodes (Bolonkin, 2008).

Modern PMS units have evolved to a level of sophisticated design that no longer requires moving parts. The central computer controls and monitors system performance. The ion thruster then processes the propellant and power to perform work. Modern ion thrusters are capable of propelling a spacecraft up to speed of 90,000 meters per second. Although, this is theoretically a feasible drive for the spacecraft, its technology is still very far away. Instead, in the following sections, the prospect of using nuclear energy for direct propulsion will be considered by heating the propellant to very high temperatures and then exiting the propellant through a standard convergent – divergent nozzle, in order to reach required velocities for interplanetary travel.

6. USING NUCLEAR REACTORS FOR PROPULSION IN SPACECRAFT

6.1 Reasons for Using Nuclear Reactors in Spacecraft Propulsion

The usage of a nuclear reactor as a source of power can be instrumental in the operation of a spacecraft. Although, nuclear propulsion can also be utilized for propelling the rocket away from Earth in a take off (so that it can escape the Earth's gravity in the atmosphere); this is not considered as a safe and practical application. In fact, in the programs conducted by the Soviets and the Americans, many unfortunate incidents took place, which caused serious problems during both the launch stage and in the atmospheric landing phase. In fact, an accident of a Russian nuclear satellite caused severe radioactive fallout over the atmosphere of Canada. This major incident was enough to persuade the United Nations to place restrictions on the usage of nuclear rockets in an atmospheric flight. (IAEA, 2005)

Hence, due to these safety concerns and restrictions, the main technique for employing nuclear reactors in spacecraft is the usage of nuclear reactors for propulsion in microgravity environment. Microgravity environment can be defined as the non – atmospheric space beyond the Earth's gravity field. In microgravity, vacuum conditions prevail and it is assumed that there will be no pressure and gravity (negligible gravity gradients are allowed) forces acting upon the structure of the spacecraft.

As mentioned in the introductory phase, using a nuclear reactor as a means of propulsion in spacecraft is an efficient way to achieve high specific impulses due to the high temperatures obtained. This stems from the thermodynamic equations presented in Section 4, which clearly demonstrates the importance of having a high chamber temperature T_c that can influence the exit velocity V_e. This is extremely significant, as many long-range missions will require having large specific impulses such as 5000 seconds or above in order to realize the scope and the duration of the mission. These specific impulses can only be reached through high temperatures that are in the order of thousands of Kelvin and obviously, it is not possible to reach these temperatures by using natural chemical burning processes and other classical methods (Czysz, 2006).

There are different approaches that can be taken for creating thrust through nuclear means. The type of reactor that is used can change depending upon the mission requirements, as well as the power conversion cycle that may be chosen.

Some approaches include:

1) Using Solid Core Nuclear Reactors as a Means for Space Propulsion
2) Using Liquid Cooled (such as Liquid Sodium) Reactors for Space Propulsion
3) Using Gaseous Reactors for Space Propulsion
    a) Open Cycle Gaseous Reactors for Spacecraft
    b) Closed Cycle (Nuclear Light Bulb) Reactors for Spacecraft

6.2 The Nuclear Physics of Nuclear Reactors in Spacecraft

The main operating principle behind a nuclear propelled spacecraft is that the propellant is transferred or pumped in to the heat generating region of the reactor. Then the fissionable fuel is pumped in to the heat-generating region simultaneously, so that the state of the nuclear criticality is reached. Once this state of nuclear criticality is reached, the fission reaction starts to occur in the gaseous mixture within the core (Czysz, 2006).

The rest of the important parameters in the fission reaction are approximately the same as the fission reaction that was first hypothesized by Enrico Fermi in December 2, 1942 (Bussard, 1965). In general, the thermodynamics of the gaseous nuclear core reactors are about the same as normal reactors defined by the fission reaction. In general, about 1 / 2 kg of U^{235} produce an equivalent amount of energy that can be produced by 3 million kilograms of fossil fuel (Czysz, 2006).

It is the neutrons as seen in Figure 6.1 that play a major role in initiating the fission reaction. The reason why neutrons are so effective is because they are free to wander at will (due to their neutral charge) and they interact through inter-nucleon forces, only when they are very close to the nucleus. Another important parameter of nuclear reactor cores in spacecraft is the fission neutrons. Fission neutrons are special neutrons that are newly born in a single fission reaction, per neutron engaged and lost in such a reaction. For natural Uranium, this number is usually on the average of 2.47

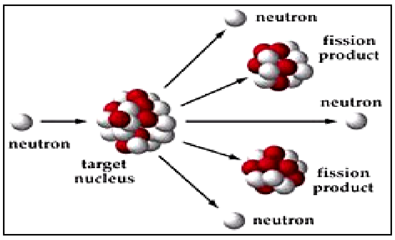


Figure 6.1: Typical Fission Reaction

In the nuclear physics of a spacecraft reactor, the absorption of some of the neutron by non-fissionable nuclei in the system is of a great concern to the design engineer. Hence, the reactor geometry is also an important consideration, since the critical size of the reactor core can be vital. In addition, the critical mass of the fuel must also be calculated carefully. In general, % 48 of the neutrons must be absorbed in the fuel for it to be a sustainable reaction (Gribbin, 1989).

In a typical fission reaction of Uranium 235, the end products would look like this:

$$n + {}^{235}_{92}U \rightarrow {}^{236}_{92}U$$

$${}^{236}_{92}U \rightarrow {}^{144}_{56}Ba + {}^{89}_{36}Kr + 3n + 177\text{MeV}$$

In the fission reaction as seen in Fig. 6.2, new fission neutrons are created in the fission and they will continue to trigger new fission reactions, as the uranium atoms are split apart to create atoms with smaller atomic weight. The exothermic reaction also produces around 177 Mev of energy and it is this energy that is used as heat (Gribbin, 1989).

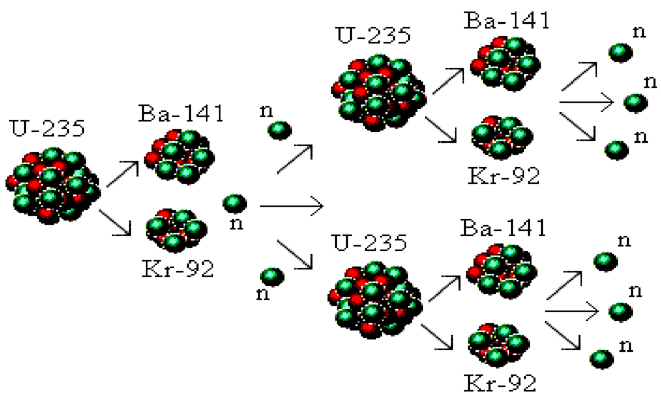


Figure 6.2 : Fission Reaction of Uranium 235

The selection of the moderator is important, as this is the best way in which the nuclear fission reaction can be sustained properly without a problem. However, since this is a spacecraft, moderators such as water cannot be used and the moderator may only be in the form of solids or in a gaseous form. It is possible to use graphite as a moderator in spacecraft, but then it introduces stability and structural problems. Hence, keeping the moderator in gaseous form is the best solution as far as the spacecraft propulsion dynamics is concerned (Turchi, 1998).

Although using fast neutrons sounds like a good idea in a gaseous reactor, using thermal neutrons is a more feasible method. The neutron flux can be maintained more easily in a gaseous chamber of a nuclear reactor in a spacecraft. In addition, the fact that the nuclear reactor is operating in a microgravity environment is also effective in several of the different nuclear reactions that take place. The neutron population in the thermal reactor has a Maxwellian distribution, which is reduced, as epithermal neutrons are introduced in to the system as dependent on the temperature flux (Turchi, 1998).

Hence, the nuclear physics aspect of this nuclear reactor is actually very straight forward as the fission neutrons bombard the fissile fuel to create a fission reaction. The presence of the thermal neutrons as seen in Figure 6.3 to be in proportion to the fission neutrons is an essential requirement (Czysz, 2006).

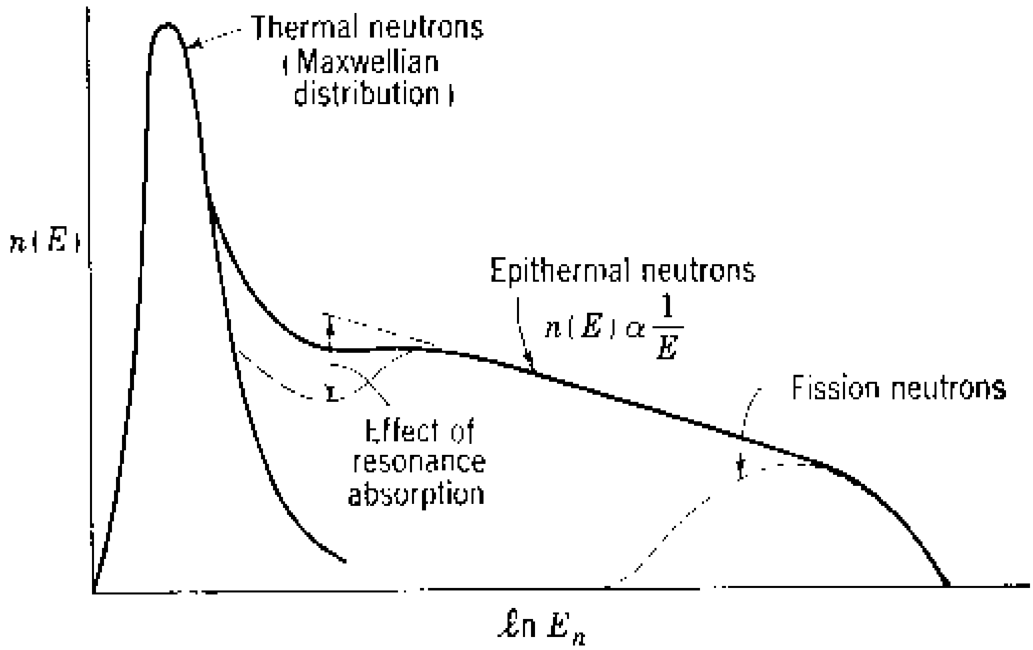


Fig 6.3: Neutron Distribution in the Reactor (Bussard, 1965)

In essence, it is the concept of breaking the strong nuclear forces that causes so much energy to be released in a fission reaction. Strong nuclear forces binds the particles in the nucleus together. Strong nuclear force is much more powerful then the repulsive forces. In a fission reaction, these nuclear forces are broken down as the atom's nucleus is fragmented after a collision from a neutron.

As a consequence, the fissile fragments are ejected and there is a corresponding mass decrease between the original atom that was fissioned and it's by products. This mass decrease is the caused by the mass that was converted into energy. The breaking of the strong nuclear force also plays some part in the total energy output of the fission reaction as well as the weak nuclear force. The mass is converted in to energy per Einstein's famous energy equation (Gribbin, 1989).

For nuclear reactions of any sort, the total numbers of protons and the neutrons are separately conserved and the rest energy state remains constant. The change in the total energy is due to changes in the binding energy from the initial to the final state. If the magnitude of the total binding energy is increased due to the nuclear reaction, the final state will be more stable than the initial state (Gribbin, 1989).

Fission process is accomplished more easily if the initial nucleus is on of those, which is already heavy enough to be naturally unstable. That is why Uranium and Plutonium as depicted in the Figure 6.4 below are the preferred fuels in any nuclear reaction. The usual choice of nuclear fuel is Uranium 235 due to its ready availability. It is the general choice for many nuclear reactors for terrestrial, naval, and aerospace applications (Czysz, 2006).

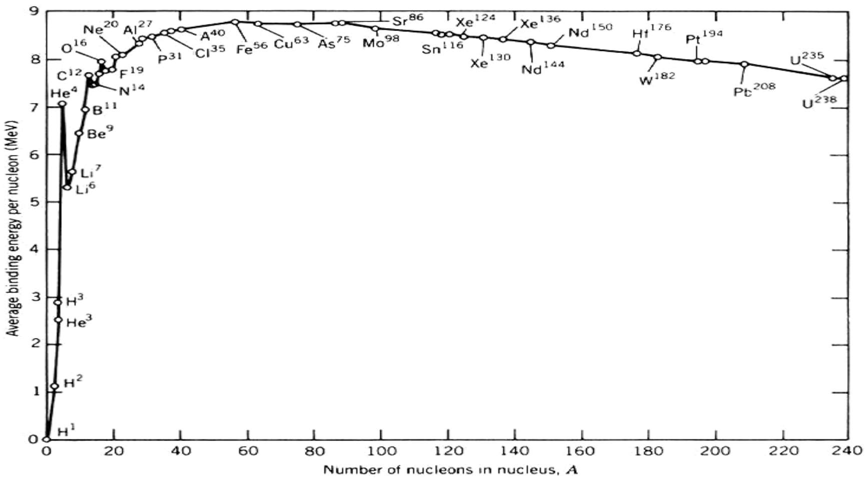


Figure 6.4 : Binding Energy as Function of Nucleons

6.3 Kinetic Energy Concepts with Nuclear Propulsion of Spacecraft

As mentioned in the preceding section, the fission process causes energy to be released by converting nuclear fuel mass into energy. Once a fission reaction occurs, the fissionable fuel atoms are divided, so that both neutrons and other high-energy particles are emitted from the nuclear fuel.

However, the mass of the particles at the end of the fission reaction is smaller than the initial mass of the particles. There is a specific difference in mass, in the beginning of the reaction as well as at the end of the reaction. This difference is expressed as energy and it is this energy expression that causes the nuclear reaction to produce energy. Hence, for spacecraft working in a microgravity environment, the difference is expressed as the energy that is used for the propulsion.

The difference in this lost mass that is converted into energy is expressed as α. In essence, this α is the mass difference that is changed in to kinetic energy. Consequently, this energy differential is converted into kinetic energy in accordance with the Einstein's famous energy equation of

$$E = mc^2 \qquad \textbf{(6.1)}$$

The processes for energy exchange are fairly easy to measure and understand. The energy released is quite large since c represents the speed of light in this equation.

$c = 3 \times 10^8$ m / sec

Therefore, the energy that is released is quite large, as compared to other forms of energy. Relativistically speaking, the mass that is lost corresponds to a decrease of the potential energy of the nuclear force that is binding the neutrons and protons. Although in Newton mechanics, both energy and mass are two different quantities; in high speeds, this equation is to be utilized to understand this process.

In essence, it is the relativistic mass that is conserved as a sum of relativistic physics:

$$mc^2 + KE \qquad \textbf{(6.2)}$$

In the above equation, m is the relativistic mass, while the rest mass is defined as m_o. By using Einstein's relativistic equations, the relativistic mass can be defined as the equation 6.3:

$$m = \frac{m_o}{\sqrt{1 - \left(\frac{V}{c}\right)^2}} \quad \textbf{(6.3)}$$

The fission reaction allows the potential energy of the nuclear force of the atom in the Kinetic Energy of the fragments. The energy that is released is in the order of 10^{13} J/kg.

Hence, the potential energy in mass m of fuel is defined as the fraction α * mc^2

$$\text{Fuel PE} = \alpha m_{fuel} c^2 \quad \textbf{(6.4)}$$

The effect of the fission is that the potential energy of the nuclear force (which binds the nucleons together) is converted into the kinetic energy of the fragments such as the nuclides, neutrons, and photons.

The kinetic energy of the fragments through collision converts the internal energy of a fluid or propellant that is present as a mass m_p. This internal energy of the propellant becomes the orderly motion of particles ejected at the speed of V_e.

In order to calculate the ideal velocity V that is reached by a mass of m_p of propellant after (α m) mass of fuel fissions, it is essential to write a relativistic energy balance equation. For purposes of simplicity, the Kinetic Energy is approximated with the expression:

$$KE = \frac{1}{2} m V^2 \quad \textbf{(6.5)}$$

Here the neutrino and the photon energies are discarded for simplicity.

Thus, the equation can be constructed by using the expressions and the variables described above by:

$$m_o c^2 = (1-\alpha) m_o c^2 + \frac{1}{2} \frac{m_o (1-\alpha) V^2}{\sqrt{1-\frac{V^2}{c^2}}} + \frac{1}{2} \frac{M_{po} V^2}{\sqrt{1-\frac{V^2}{c^2}}} \quad \textbf{(6.6)}$$

where:

m_o = fuel mass

M_{po}= propellant mass as rest

α = The mass of the propellant that is converted into energy by fission

c= speed of light

V = exhaust velocity

This equation can be rearranged, so that it is possible to see a more clear result. Hence:

$$m_o c^2 - (1-\alpha) m_o c^2 = \frac{1}{2} \frac{m_o (1-\alpha) V^2}{\sqrt{1 - \frac{V^2}{c^2}}} + \frac{1}{2} \frac{M_{po} V^2}{\sqrt{1 - \frac{V^2}{c^2}}} \qquad \textbf{(6.7)}$$

This simplifies into:

$$\alpha m_o c^2 = \frac{1}{2} \frac{m_o (1-\alpha) V^2}{\sqrt{1 - \frac{V^2}{c^2}}} + \frac{1}{2} \frac{M_{po} V^2}{\sqrt{1 - \frac{V^2}{c^2}}} \qquad \textbf{(6.8)}$$

By rearranging the last part of the equation and by simplifying:

$$\alpha m_o c^2 = \frac{1}{2} \left[\frac{m_o (1-\alpha) V^2 + M_{po} V^2}{\sqrt{1 - \frac{V^2}{c^2}}} \right] \qquad \textbf{(6.9)}$$

Then it is necessary to continue to rewrite this equation to get it into a more usable form. The purpose in making all of these arrangements is to get it into a form where the speed is the dominant form in the left side of the equation:

$$\frac{V^2}{c^2} = \frac{2}{\sqrt{1 + \frac{2}{A}} + 1} \qquad \textbf{(6.10)}$$

Where A is defined as:

$$A \equiv \frac{2\alpha^2}{(1-\alpha)^2\left(+\dfrac{M_{\mathrm{po}}}{m_{\mathrm{o}}(1-\alpha)}\right)} \tag{6.10a}$$

Therefore, now there is an expression for the change in the exhaust velocity of the spacecraft by using the mass ratio of M_{po}/m_o, as well as by using the α value, which is the amount of mass that is fissioned. This equation gives a way to examine the performance of the spacecraft.

There are several important things that can be noted in this equation. For example, if the limit of α going to 1 is calculated, then it can quickly be seen that the velocity of the spacecraft will approach c, which is the speed of light.

The typical value of α for Uranium 235 fission reaction can be given as 9.1×10^{-4}. It is interesting to note that as the propellant M_p is added, then both the velocity and the specific impulse drops down rapidly. But for the reactor to work at reasonable temperatures and also in order for it to produce reasonable thrust, propellant must be added.

Thus, this will allow getting a mathematical relationship between the amount of material that is present for the fission, as well as the velocity that is attained. It is also possible to see a relation between the amount of fissionable material and specific impulse by using this method and the specific impulse equation 4.1

The equation defined above in (6.10), can be combined with the specific impulse equation in (4.1). Then the approximation for Specific Impulse in Nuclear Spacecraft becomes:

$$I_{sp} = \frac{\sqrt{\dfrac{2c^2}{\sqrt{1+\dfrac{2}{A}}+1}}}{g} \tag{6.11}$$

where:

$$A \equiv \frac{2\alpha^2}{(1-\alpha)^2\left(+\dfrac{M_{\mathrm{po}}}{m_{\mathrm{o}}(1-\alpha)}\right)}$$

The velocity differential of the spacecraft can be graphed as a function of the mass of material that has been fissioned (alpha). This also shows mathematical relationship between the amount of material that fissions as well as the velocity of the spacecraft. Thus, this gives an understanding between the amount of fission that occurs and the kinetic energy that is imparted into the spacecraft (which is translated in to velocity components) as a result of these fissions.

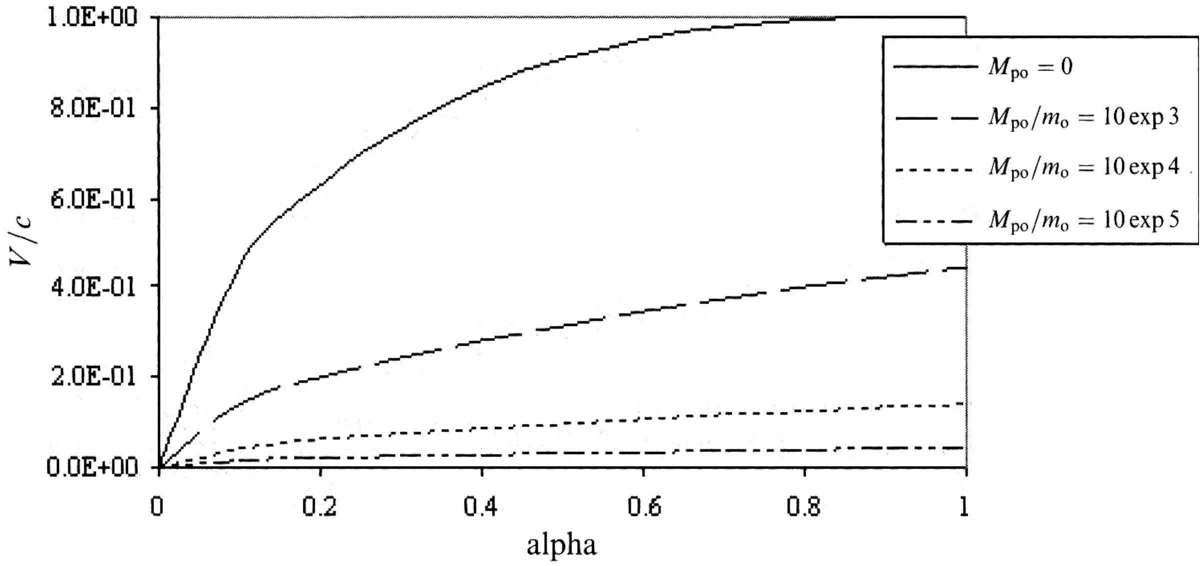


Fig 6.5: Velocity as a Function of Fissioned Fuel Mass Alpha

Hence, a lower specific impulse is a technical necessity. It is important to note that in Fig. 6.5, if $\alpha = 1$, then this can be described as matter and antimatter annihilation and the theoretical speed limit reaches and becomes the speed of light c. If the special case of $M_p=0$ is analyzed, it is seen that this means that all of the energy developed by the fission reaction ends up as kinetic energy of the fragments. The work point of the engine is on the upper curve and V or I_{sp} (specific impulse) is maximum for a given α.

This means that the fission products are the propellant and they are ejected (as they are) with all of their kinetic energy. Moreover, the fission products are perfectly collimated and full energy exchange occurs. In this method, the thrust is modest. In addition, the mass of fuel fissioning per unit time is naturally low and thus a specific impulse of 10^5 sec can produce a thrust on the order of 1000 Newton (Czysz, 2006).

Understanding the kinetic energy concepts of a nuclear reactor allows the comprehension of the fission process as well as the velocity differential of the spacecraft. Especially for long-range spacecraft of the future, partial light speed velocities (such as $1/10^{th}$ of light speed) will be feasible and also necessary in order to reach longer distances. Hence, the analysis for orbital maneuvers for the spacecraft can be made more accurate with the information presented in the above section.

6.4 Using Gas Core Nuclear Reactors for Propulsion in Spacecraft

In gas core nuclear reactors, the nuclear reactor is used as a means of generating heat, so that the propellant in the spacecraft (usually liquid hydrogen) can be heated up in the core and be thermodynamically expanded into a gaseous / plasma state. Once this occurs, the high temperature plasma propellant will be discharged from a thermodynamic nozzle. This discharge will cause the spacecraft to accelerate and to gain momentum. As the temperature of the propellant increases; this will cause a higher acceleration rate and hence a higher specific impulse (Turchi, 1998).

Thus, it is essential to reach as high temperatures as possible in the reactor core, in order to make sure that the propellant can be heated up in a maximum way. As the temperature of the propellant goes up, so does the speed of the spacecraft, since the propellant is expanded more forcefully, to create a more powerful thrust and a more speedier exhaust velocity. However, the design of the nuclear reactor must be done in such a way to make sure that the necessary temperature gradients can be reached with minimum mass and with minimum dynamic complications that will take place (Czysz, 2006).

One of most useful models for nuclear reactors in spacecraft is without a doubt “gas core nuclear reactor” in spacecraft. These types of nuclear reactors are easier to design and more practical to utilize, due to the fact that the outer shell of the nuclear reactor will not be penalized by the high temperature output of the nuclear reactor. Hence, unlike a solid core nuclear reactor, it’s possible to have nuclear reactor temperatures of above 5000 K, since high temperatures will be needed for propulsion. Also, unlike solid core reactors as seen in Fig 6.6, the designer won’t be limited in the choice of materials, as the engineer will easily be able to use a larger range of materials since the whole operation will take place in the gas core of the reactor.

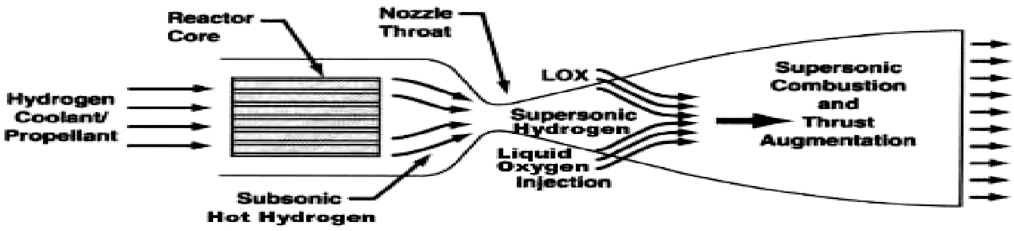


Fig 6.6: The Usage of a Nuclear Reactor for Heating the Propellant in Spacecraft

In essence, the simplest form of a nuclear gas core reactor for rockets is using an open chamber as a means to have a nuclear reaction. Here, the core is open to the thermodynamic nozzle, the propellant (such as hydrogen) is passed through the open core geometry, and it is allowed to exit through the nozzle as it heats up and expands.

The propellant also undergoes a severe thermodynamic transformation as it is taken from a liquid state (around 27 Kelvin) in to gaseous and plasma state (around 5000 Kelvin degrees) very quickly, while it passes from the core of the open cycle gaseous core nuclear reactor in the spacecraft. Then, the expanded and the heated gas is discharged through the nozzle to create the necessary thrust momentum and the required exhaust velocity as seen in Fig. 6.7 (Czysz, 2006).

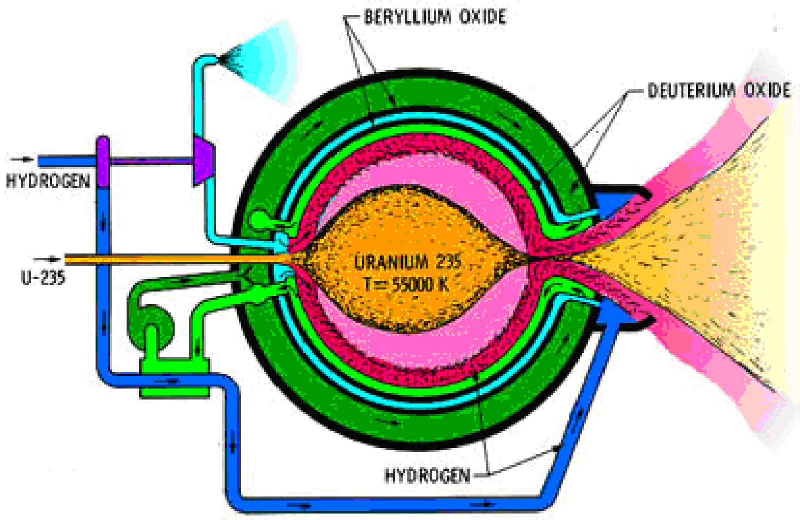


Fig 6.7: Fictional Representation of a Gas Core Nuclear Rocket

The advantage in this model is that it is relatively simple, as both the nuclear reaction as well as the heating of the propellant takes place at the same geometrical location. There is no need for a fancy thermodynamic nozzle and there is no need for extreme heat transfer and heat exchangers, which can complicate the process. However, the main problem with these types of open core gaseous nuclear reactors is the fact that some fissile material will be discharged during the exhaust phase and thus this can cause a radioactive danger and also the loss of radioactive material (Czysz, 2006).

However, in a closed cycle model of gas core nuclear rockets, the nuclear reaction is semi contained in a chamber. Hence, the heat that is generated in the closed cycle nuclear chamber is transferred into the propellant that is passing from the outer surface of the closed chamber. The heat transfer is quite simple, as no complex heat transfer mechanisms are used (Turchi, 1998).

In essence, the nuclear reactor again heats the propellant and as a result, the propellant is discharged from the nozzle to create the necessary exhaust velocity. These types of closed cycle gaseous core reactors are called the "*Nuclear Lightbulb Model*". This model as seen in Fig. 6.8, has been preferred by the Americans in the 1960's for its ability to produce continuous pulse of energy in a steady state, but it was discontinued due to heavy demands for materials technology in its design (Williams, 1997).

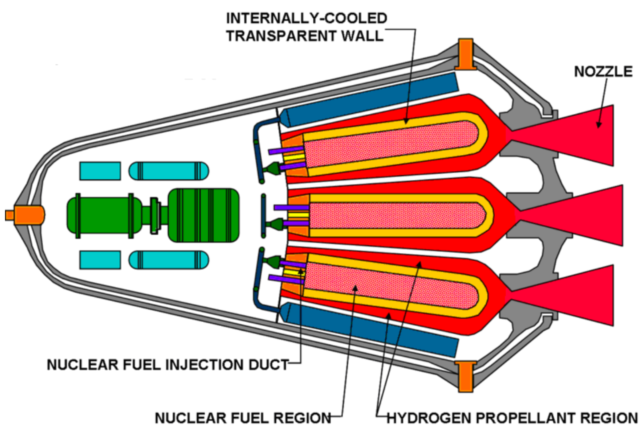


Fig 6.8: Representation of Nuclear Lightbulb Model

The advantage with this model is the fact that, fissile material is contained in the closed core geometry and as a result, the productivity is higher. The disadvantage of this type of a nuclear reactor in spacecraft is the fact that it can yield a lower specific impulse (Williams, 1997).

Moreover, it can have a lower thrust ratio due to the fact that higher amounts of mass will need to be involved due to the shielding etc. However, some of the design problems associated with using nuclear reactors for propulsion can be easily handled by using the nuclear light bulb model in the spacecraft nuclear reactor design.

As another added alternative, when the nuclear light bulb model is used as a method to generate electricity, it can be combined with photovoltaic systems. This method also does not involve the release of any radioactive material from the rocket, unlike other designs, which would cause nuclear fallout if used in a planetary atmosphere (Williams, 1997).

7. THE PRINCIPLES OF GASEOUS CORE REACTORS FOR ROCKETS

7.1 Using Nuclear Reactors for Heating Hydrogen as a Propellant

In this master's degree thesis, a gaseous core nuclear reactor, which has an open chamber, will be analyzed. The gaseous core nuclear reactor has an open core and the nuclear fuel inside is in gaseous form. The fission reaction also takes place in a gaseous setting. The hydrogen will be pumped into the open chamber of the gaseous core nuclear reactor, so that it can get super heated. The function of the nuclear reactor core will be to intake hydrogen molecules, which will serve as a fuel for the thrust mechanism. Hence, this space vehicle will in fact function with two different fuels. The first fuel of uranium will be for the nuclear reactor and for sustaining the nuclear reaction. The second fuel will be the hydrogen itself, as it will serve as a propellant for the rocket (Czysz, 2006).

The hydrogen will enter the gaseous core reactor chamber in subsonic speeds. Then this hydrogen which enters the reactor core in subsonic speeds will be super heated to high temperatures of 5000 K to 10,000 K depending upon the design parameters. Once hydrogen is heated to these temperatures, it will also reach supersonic speeds due to the excitation of its molecules (Czysz, 2006).

As soon as supersonic speeds are reached, the hydrogen will be expelled out from the exhaust of the rocket in a plasma form. Once this exhaust takes place, the rocket will have a forward momentum in a microgravity environment and as a result, the rocket will have momentum exchange in accordance with the Newton's Third Law.

7.2 The Essentials of Nuclear Fuels for Gaseous Core Nuclear Reactors

The concept of using a nuclear reactor to heat up the hydrogen propellant seems to be a technologically sound idea. As stated many times before in this thesis, the main criterion for judging a spacecraft's performance is the specific impulse of the rocket (or the spacecraft).

As the specific impulse of the spacecraft gets bigger, the distance that the spacecraft can travel can increase greatly. Thus, when designing a spacecraft that uses a nuclear reactor for propulsion; it's important to design the specific impulse to be as high as possible (Czysz, 2006).

The main design problem for using a nuclear reactor for propulsion is the control of its temperature. As a background, it's essential to analyze the potential of a fission reaction in more detail by analyzing the associated temperatures that can be reached if uranium fission products are utilized to heat the hydrogen (as a propellant fuel) for exhaust through a rocket nozzle. Consequently, hydrogen provides the highest specific impulse in all elements for any given temperature. Therefore, as it can be seen from the preceding information, temperature is an important factor for the functioning of the spacecraft, as well as the factors such as its velocity and the maximum distance that it can travel.

Moreover, it is possible to state that the specific impulse and the corresponding gas temperature may be plotted as function of dilution ratio and also of burn up fraction. Dilution ratio is a very important factor, as it is defined as the ratio of the hydrogen flow rate to the uranium fission rate. Also, burn up fraction is the ratio of uranium fission value to the total uranium that has been utilized. These factors are defined as (Grayamov, 1990):

Dilution Ratio (DR): $\dfrac{\text{Hydrogen Flow Rate}}{\text{Uranium Fission Rate}}$

Burn-up Fraction (BF): $\dfrac{\text{The amount of uranium fissioned}}{\text{Total amount of uranium available}}$

So, as the dilution rate nears the value of zero, the propellant gas (hydrogen) can be thought of as having pure fission products inside the stream. Moreover, it can be seen that the specific impulse at this stage reaches around 10^6 seconds value. As a result, the corresponding temperature will be very high. However, the maximum allowable temperature will be limited by the confinement strength of the material that is used in the construction of the nuclear reactor in the spacecraft (Czysz, 2006).

One way to increase the limitation temperature of the nuclear rocket materials is by utilizing fissionable material in its gaseous state. The hydrogen, which is kept in a liquid state, will then be converted into a gaseous form. Hence, either this gaseous form of hydrogen in a fluid state can be mixed directly with the fissioning uranium or it can be heated by using the thermal radiation from the uranium that is in a fission process (Grayamov, 1990).

It is possible to construct a gaseous core nuclear rocket by using a gaseous mass of fission material (such as Uranium Hexafluoride or Uranium Tetrafluoride) and then using a reflector – moderator to contain the nuclear reaction. It could be possible to use both Uranium 238 as well as Uranium 235 with different moderators (Turchi, 1998).

In an example, if 17,000 degrees Celsius needs to be reached for the required specific impulse and if a uranium pressure of 70 atmospheres must be attained with 45 cm thickness in the moderator, then reactor diameter can be calculated. Roughly, calculations can show that the required reactor diameter will be around 1 meter as seen in Fig. 7.1 (Turchi, 1998).

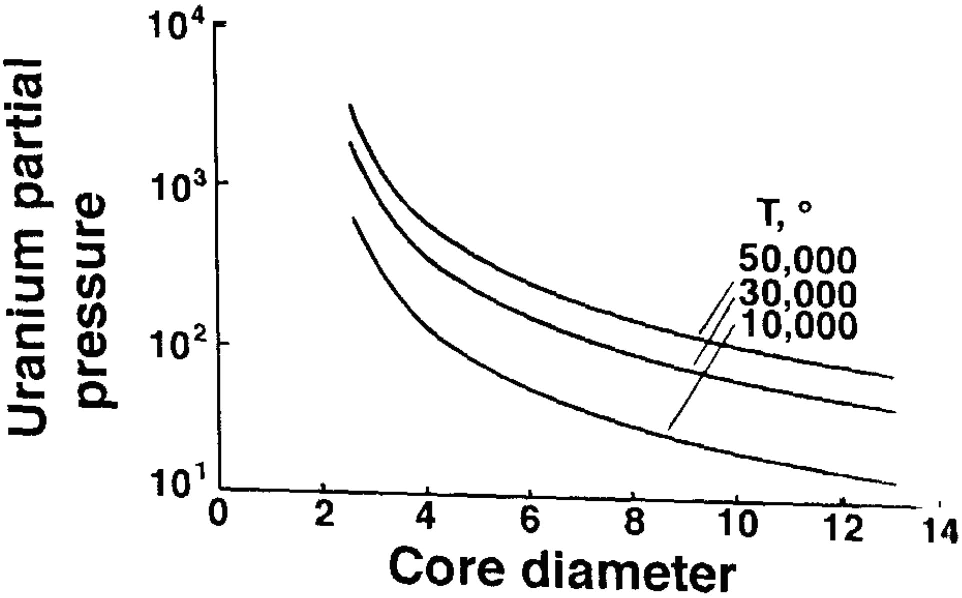


Fig 7.1: Gaseous Core Reactor U235 with D2O Reflector (Turchi, 1998)

In order to reduce the consumption of the uranium; it is possible to use the hydrogen as a diluent. This way, the hydrogen is utilized to reduce the uranium partial pressure. It is imperative to maintain a total pressure value of 68 - 70 atm as much as possible (Grayamov, 1990).

After several studies, it can be suggested that the best possible uranium consumption rate is %1 of the hydrogen consumption rate. This is a good ratio to help the aerospace engineers and the nuclear engineers in designing a space vehicle with a nuclear reactor (Turchi 1998).

Hence as a practical calculation for a sample space mission:

The total amount of hydrogen propellant fuel: 100,000 kg

The total amount of U235 fission fuel needed: 1000 kg

In order to have a reasonable and usable diameter value in the nuclear reactor, it is important to increase the uranium density without increasing the overall pressure. Also, it is essential to note that the flow rate of the uranium must be kept relatively the same as the hydrogen flow rate. It is important to realize that the hydrogen atoms must be diffused through uranium atoms, so that the velocity difference is uniform. If there is non uniform mixture of hydrogen and uranium, this will cause uneven heating of the hydrogen and uranium plasma. This in turn can create chaotic and turbulent flow within the spacecraft propulsion system (Grayamov, 1990).

In fact, in order to create a stable system in the nuclear reactor, the velocity of the uranium fuel must be less than 1/1000 of the velocity of the hydrogen propellant fuel. This is the best way in which an equilibrium can be reached in the nuclear reactor as seen in Fig. 7.2. Moreover, for separation ratio constraints, it is essential for the nuclear reactor to have a slower speed of the Uranium gas, since this will mean that lesser uranium fuel will exit the spacecraft through the nozzle (Turchi, 1998).

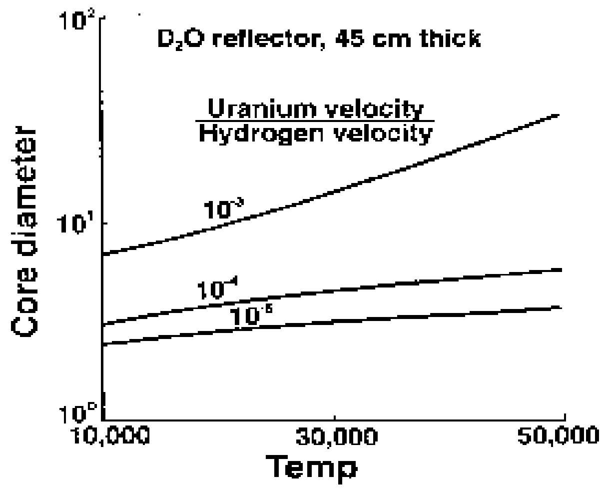


Fig 7.2 : The Propellant Ratio for Gaseous Core Reactor for U235 (Turchi, 1998)

7.3 Optimal Design Parameters for Gaseous Nuclear Reactors in Spacecraft

The principals of designing a gaseous nuclear reactor have been established with the parameters stated in the above section. It is essential to create certain conditions, in order to heat the hydrogen, so that it can be exited from spacecraft's nozzle.

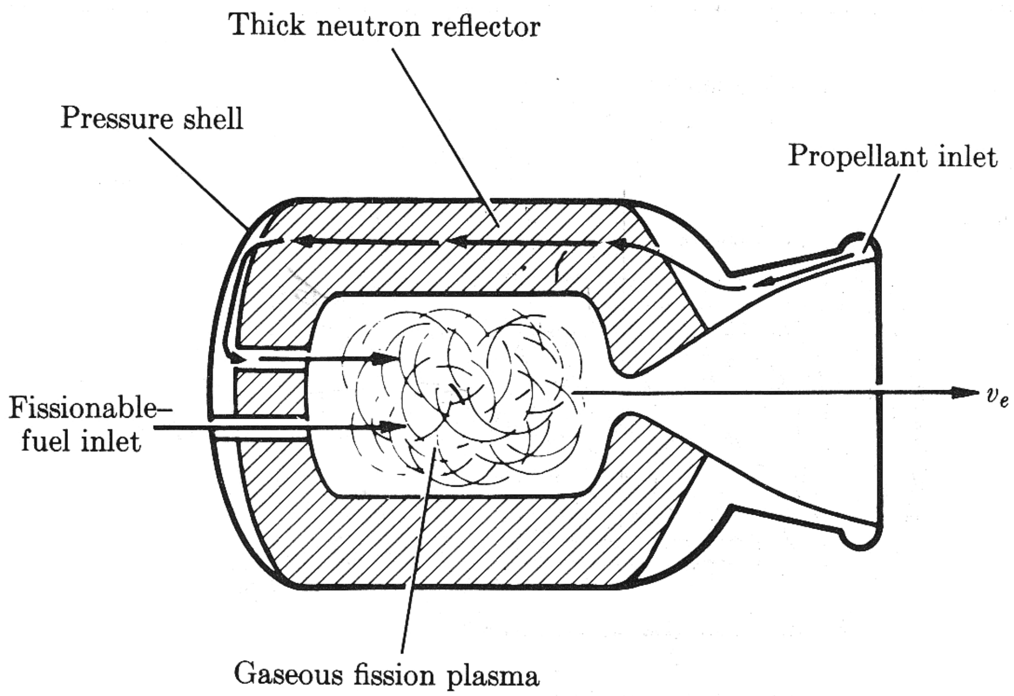


Figure 7.3 : Design of a Gaseous Core for a Nuclear Rocket (Bussard, 1965)

In essence, an open chambered gaseous nuclear core powered rocket is quite simple in its design as seen in Fig. 7.3. There is a need for two different inlets in to the core chamber. One inlet is comprised of the propellant (preferably H_2) and the other inlet is comprised of fissionable fuel (such as Uranium derivatives). The propellant inlet can also eject the propellant in liquid form, but it will turn into a gaseous state as soon as it enters the reactor core due to the high temperatures involved there. Most experiments show that hydrogen will become gaseous as soon as it enters the first 2-3 cm of the reactor core (Rom, 1962).

However, the nuclear fuel inlet needs to eject the nuclear fuel in gaseous form. The ejected nuclear fuel will have its own velocity and it will start to fission as it comes in contact with the fission neutrons. The process will generate heat. As a result, the propellant gas, which is circulating in the reactor core, will be heated up to very high temperatures due to the fission reaction. Then by magnetic and thermodynamic means, the propellant will be ejected with high speed from the nozzle. The reverse reaction according to the Newton's Law of Motion will cause the spacecraft to move forward, as the propellant particles are ejected at the rear at high speeds (Rom, 1962).

The initial conditions for the design of the nuclear reactor with a gaseous chamber in a spacecraft can be summarized as:

-<u>Type of Reactor</u>:	Gaseous Nuclear reactor
-<u>Type of Reaction</u>:	Simple Uranium Fission Reaction
- <u>Preferred Fuel</u>:	Uranium 235 (U_{235})
- <u>The Form of the Fuel</u>:	Gaseous
- <u>Preferred Propellant</u>:	Hydrogen (H_2)
- <u>Form of Propellant</u>:	Gaseous
- <u>U235 Consumption Rate</u>:	%1
- <u>U235 Flow Rate</u>:	100:1
- <u>Standard U235 Pressure</u>:	68 atm = 70.3 kg force / square centimeter
- <u>Required Core Diameter</u>:	Between 1 meter to 3 meters (1m < D > 3m)
- <u>D2O Reflector Thickness</u>:	Between 35 cm to 50 cm
- <u>Required Uranium Velocity</u>:	1/1000 of the hydrogen velocity

Hence, in a proper design of a gaseous nuclear reactor for a spacecraft, the above conditions must be satisfied before further design parameters can be introduced. Thus as a summary it can be stated that:

1) There should be 100 times more hydrogen injected into the system as compared to the amount of uranium that is present.
2) The U235 pressure must be kept around 68 atm for best results.
3) In order to make sure that the spacecraft does not have more mass than it is required, it is essential to limit the core diameter to less than 3 meters for best results. As the core diameter gets bigger, the space that can be allocated for the crew of the spacecraft as well as the payload of the spacecraft will be reduced considerably. Moreover, as the diameter of the core gets bigger, then

new nuclear control instability problems will be introduced in to the spacecraft.

4) Again, for mass limitation purposes, the reflector shield thickness should not be more than 50 cm. As the deflector shield gets thicker, this will introduce many structural design problems to the spacecraft and thus the spacecraft may not be able to carry a proper payload. Although the spacecraft does not have weight in space, proper attitude control as well as proper momentum transfer and diffusion depends on the mass distribution of the spacecraft (Mohler, 1961).

5) For proper heat transfer and also for flow control purposes, the velocity of the uranium must be kept as less than 1/1000 of the velocity of the hydrogen flow (Mohler, 1961).

6) It may be required to use a more corrosive form of Uranium for stability control problems. Some research indicates that the best form of uranium for such a reactor and also for such a spacecraft can be to use gaseous uranium hexafluoride. This will introduce corrosion containment problems to the nuclear reactor and also to the spacecraft. However, by using anti corrosion techniques to treat the material, this can be overcome (Turchi, 1998).

7) The specific impulse of the spacecraft must be kept as a function of propellant temperature that is attained in the nuclear reactor core. In fact, the specific impulse equation can be written as the inverse of the square root of the enthalpy of the propellant (hydrogen). Hence, as higher temperatures are reached in the "hydrogen-uranium" slush mixture, higher specific impulses can be reached. (Turchi, 1998)

8) It is important to remember that at such high temperatures of 20,000 degrees Celsius to 50,000 degrees Celsius, the uranium mixture is partially ionized due to the temperature. However, due to the molecular inert nature of the hydrogen, the hydrogen propellant does not easily become ionized. Hence, this means that the nuclear fuel or the Uranium is ionized, while the propellant fuel or the Hydrogen is not ionized. This can lead to regulating the flow of the "hydrogen-uranium" mixture by using a magnetic field in order to contain the diffusion forces (Bruno, 2006).

9) As a strong field is applied to the mixture, the maximum permissible hydrogen flow rate is determined. As the applied field gets stronger, the hydrogen flow rate through the ionized uranium becomes greater (Sanchez, 2005).

10) It is essential to keep the flow of hydrogen in a steady state for controlling the fission / combustion reaction. Since hydrogen will enter the reactor in liquid form, it will condense after it enters the reactor core due to the high thermal heat. However, this condensation in the first few centimeters of the spacecraft can cause problems, as a mini standing wave can occur due to high-speed change in the phase (Sanders, 1963).

11) The method of injecting the uranium is also important for stability, as the reactivity of the reactor can be affected if there are variations in the uranium cloud that is present in the reactor chamber. It is essential to keep the uranium gas cloud in a uniform and steady state throughout the nuclear reactor chamber (Sanders, 1963).

12) Once the diffusion forces are contained, then the super heated hydrogen can be exited from the nozzle of the spacecraft in order to create a high profile velocity with a high specific impulse (Mohler, 1964).

13) Thermal radiators must be used so that the excess heat can be dissipated into space. Special care must be given to the nozzles of the spacecraft, as they will be susceptible to heat damage (Taylor, 2009).

14) Vortex field flow may need to be investigated, so that the hydrodynamic flow stability problems can be solved during the design stage. Radial flow for each vortex strength must be determined for stability analysis (Mohler, 1964).

15) Fission fragment injection must also need to be contained for proper use of the nuclear fuel for financial purposes.

16) The Separation ratio must be small enough, so that most of the nuclear fuel is retained within the system.

Besides the nuclear reactor problems mentioned above, it is important to stress the fact that the thermodynamic range and the performance of the spacecraft must also be maintained. Thus, it can be stated again that the range of the spacecraft is proportional to the propellant chamber temperature.

This is a important design concept that must be kept foremost, while the nuclear reactor is designed for the spacecraft. As the nuclear reaction progresses, the temperature of the reactor chamber will increase the temperature of the propellant in the chamber that is present in the vortex flow (Ellerbrock, 1963).

Hence, by controlling the temperature of the chamber, it is possible to attain a direct control of the exhaust velocity of the spacecraft. It is also important to note that the amount of heat that can be generated by a nuclear reactor is not limitless, since there will be design limitations due to the materials being used in the design and in the construction of the nuclear reactor itself in the spacecraft. The reactor core outer walls must be able to control the heat or the rest of the spacecraft will have trouble retaining its integrity in the outer shell (Marjon, 2002).

If this relation is kept in mind, then the rest of the design problems mentioned above can be centered on this. Thus, the peak gas temperature in the nuclear reactor chamber along with the resulting specific impulse can be calculated. As it can be seen on the diagram below of 7.4, the low molecular weight of hydrogen makes it an excellent propellant (Heister, 1998).

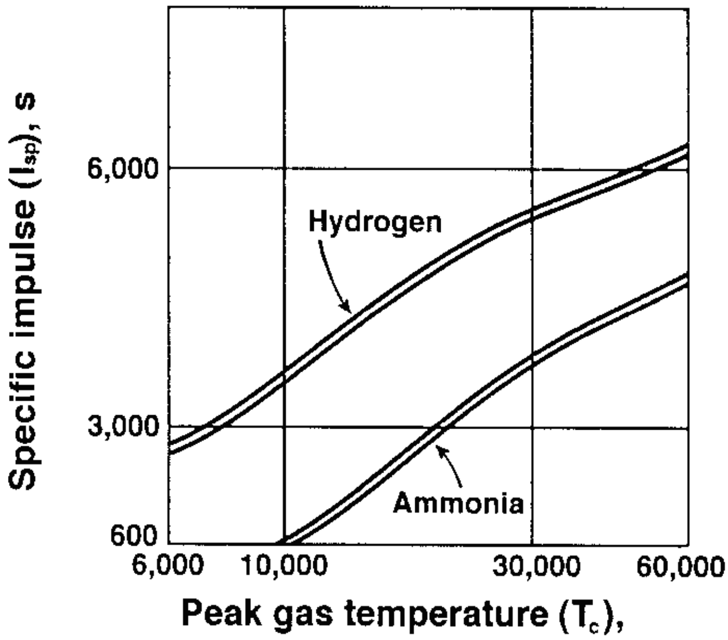


Fig 7.4: Nuclear Core Temp. of the Propellant vs Specific Impulse (Turchi, 1998)

7.4 Separation Ratio and the Flow Rate of Nuclear Fuel at Gas Core Reactors

As mentioned in the relevant sections above, both the nuclear fuel and the propellant fuel is in gaseous form. The propellant and the fissionable fuel are inputted through various inlets into the reactor core. The reactor core is shielded with a thick neutron reflector to act as a moderator and to allow the fission reaction to take place within the core.

Hence, the gaseous propellant fuel and the gaseous nuclear fuel are injected together in a turbulent flow scheme. Both the fuels are mixed and intertwined with each other. Once the reactor fission criticality is ensured through the minimum required fissionable nuclear fuel density, the fission reaction takes place in the chamber. As a result, the fission reaction heats up the propellant plasma and high temperatures are reached in the reactor core. (Bussard, 1965)

Once the mixture reaches a certain temperature, it exits the spacecraft on its rear end through an exhaust outlet. This directly works in concordance with the Newton’s Third Law, which deals with the conservation of momentum. Hence, the heavy exhausts of gases cause the spacecraft to impart momentum to the outside. As in par with momentum conservation, the spacecraft gains forward momentum in proportion to the speed of the exhaust gas (Hill, 1992).

Usually low molecular weight propellant gases are chosen for the sake of the simplicity, since the energy content per unit mass is inversely proportional to the atomic mass of the working fluid used. Thus, since combustion is not an issue in nuclear propulsion in a microgravity environment of space, it is absolutely clear that the propellants of the lowest possible atomic or molecular weight are preferred for use in temperature limited rocket motors (Hill, 1992). Due to thermodynamic principles, the propellant gas will gain more kinetic energy as its temperature is raised and the particles will vibrate faster.

Hence, the temperature of the propellant rises in proportion to the temperature of the nuclear fission reaction in the core of the spacecraft. Therefore, a more powerful fission reaction will yield higher temperatures and the spacecraft will have a higher exhaust speed (Turchi, 1998).

However, the main problem as mentioned in the above chapter is the amount of uranium that is lost in the exhaust chamber. Due to the open nature of a gaseous reactor spacecraft, both the propellant fuel and the nuclear fuel are both emitted from the spacecraft. This can be problematic, since uranium fuel is a very costly element. Recent prices of usable Uranium 235 in solid form can go up as high as $5600 per kilogram. Moreover, in order to provide it in gaseous form, it needs to be provided as Uranium Hexafluoride solution and that can be even more costly in the order of $10,000 per kilogram (Turchi, 1998).

Thus, it can be see that the cost of uranium fuel that has not undergone fission and which has been thrown out of the spacecraft without being used, can be quite high. In fact, this can cost millions of dollars per each space mission. Hence, the main problem is to make sure that the uranium fuel is somehow retained in the nuclear reactor chamber without exiting the spacecraft through the exhaust chamber.

One way to achieve this would be by using a vortex flow scheme in a gaseous heater rocket reactor. By creating a vortex flow as seen in Fig. 7.5, some of the uranium fuel can be stopped from exiting the spacecraft until it has fissioned to provide momentum to the spacecraft. One way to measure this would be by using a separation ratio to define the amount of fissionable fuel that is separated from the end product in the exhaust gas (Heister, 1998).

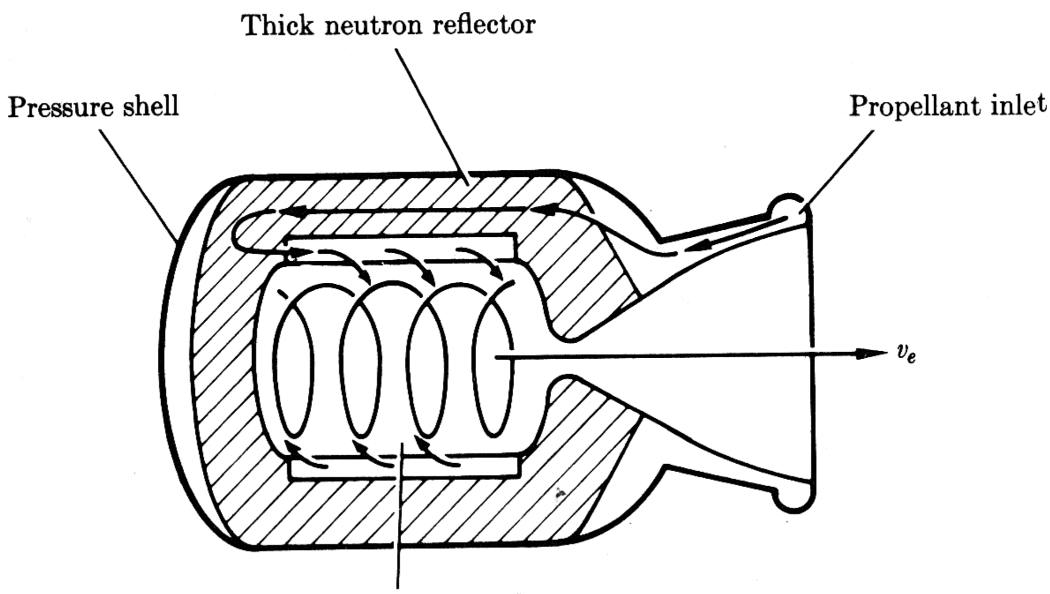


Figure 7.5 : Rotating Fuel Concept for the Separation Ratio (Bussard, 1965)

By calculating this separation ratio, there will be a valuable tool for nuclear spacecraft calculations. Moreover, by knowing the amount of fuel that is still waiting fission, it is possible to calculate the performance characteristics. This can easily lead to calculating the cost of the nuclear fuel and hence the total cost of the mission per the specific impulse that has been obtained (Kulcinski, 1997).

The best way to calculate specific impulse is by using performance equations for the spacecraft. It can be shown that the performance level and the fuel loss are in relation to each other with the following equation (Bussard, 1965):

$$\frac{P_c}{I_{sp}} = \frac{2 I_{tot} \rho_f S}{3 m_f} \tag{7.1}$$

where:

P_c = Chamber pressure in the nuclear reactor of the spacecraft (atm)

I_{sp} = Specific impulse of spacecraft (sec)

I_{tot} = Total Impulse of the spacecraft (kg.sec)

m_f = The amount of fissionable fuel that must be used (kg)

ρ_f = gaseous fuel density required for nuclear criticality of the reactor (gm/cm^3)

S = Separation Ratio (dimensionless)

Now it is possible to pull the m_f, so that the equation is rearranged:

$$m_f = \frac{2}{3} \frac{I_{tot} I_{sp} \rho_f}{P_c} S \tag{7.2}$$

The equation clearly shows that there is clearly a relation between the amount of fissionable fuel that is used and the separation ratio. For example, in a hypothetical case:

P_c = 100 (atm)

I_{sp} = 1000 (sec)

I_{tot} = 10^8 (kg.sec)

ρ_f = 10^{-3} (gm/cm^3) (general fuel density for fuel criticality)

If these are placed into the formula above, it can be seen that if the Separation Ratio = 1, then this will result in $m_f = 6.7 \times 10^5$ kg of fissionable fuel needed. Evidently, this is quite a large amount.

Hence, the separation ratio need to be quite small to get the value of m_f = 1000 kg or less. The recommended value of S should be less than $\mathbf{S < 10^{-3}}$**.** Then the values that are calculated will be workable and cost efficient for a normal long range mission. Otherwise, with separation ratios bigger then this value, the mission planner will be faced with millions of dollars worth of nuclear fuel that will need to be used in order to fly the spacecraft. Unfortunately, for most nuclear space missions, such high costs will not be tolerable and it can place the mission in jeopardy. In some instances, magnetic confinement schemes can be used to separate the ionized uranium fuel from the hydrogen fuel. Magnetic fields can be generated to suck the uranium particles from the gaseous fluid vortex formed within the nuclear reactor (Turchi, 1998).

7.5 Systems Performance Analysis of Gas Core Nuclear Reactors in Spacecraft

The specific impulse is closely tied to the exhaust velocity. The specific impulse formula can be used, to help calculate the exhaust velocity for spacecraft using a gaseous nuclear reactor core.

The specific impulse equation of 4.1 and Total Impulse Equation of 4.2 can be combined to form Equation 4.3, which can show the relation between exhaust velocity and total thrust.

$$I_{sp} = \frac{V_e}{g_c} = \frac{F}{w_{tot}} \qquad \textbf{(7.3)}$$

The parameters are:

F = Total thrust

W_{tot} = flow rate of mixture weight

The maximum possible exhaust velocity at adiabatic expansion to zero pressure can be expressed as (Bussard and Delauer, 1958):

$$V_e = \sqrt{\frac{2g_c\gamma}{\gamma - 1}\frac{P_c}{\rho_{tot}}} \qquad \textbf{(7.4)}$$

The exhaust velocity is also given from Equation 4.1 as $V_e = I_{sp}\,g_c$

Then it is possible to combine the two equations 7.4 and 4.1, in order to try to separate mixture chamber pressure as a function of specific impulse:

$$I_{sp} = \frac{\sqrt{\frac{2\ g_c\gamma}{\gamma\ -\ 1}\frac{P_c}{\rho_{tot}}}}{g_c} \qquad \textbf{(7.1)}$$

The above equation will be useful for creating a performance equation for the analysis of a nuclear spacecraft.

The above equation can be rearranged, so that some new equations that can help to determine the chamber pressure for the nuclear reactor is formed.

$$I_{sp}g_c = \sqrt{\frac{2g_c\gamma}{\lambda - 1}\frac{P_c}{\rho_{tot}}} \qquad \textbf{(7.6)}$$

With some algebraic work, the terms can be separated into more meaningful components. Then the Pressure component can be formed on one side and the others on the other side.

$$P_c = \rho_{tot}\frac{g_c(\gamma - 1)}{2\gamma}(I_{sp})^2 \qquad \textbf{(7.7)}$$

It is possible to substitute for one of the specific impulse (I_{sp}) term in the above expression with Equation 4.1:

$$P_c = \frac{g_c(\gamma - 1)}{2\gamma}\rho_{tot}(I_{sp})\ \frac{F}{w_{tot}} \qquad \textbf{(7.8)}$$

If this equation is reordered, so that it shows more sensible and workable results, then as a final result of Reactor Core Power Density:

$$P_c = \frac{g_c(\gamma - 1)}{2\gamma}(I_{sp})I_{tot}\frac{\rho_{tot}}{w_{tot}} \tag{7.9}$$

If this equation is worked further, there will be some more results, which can help to define mixture chamber pressure. This is also an important expression that can help to define the necessary parameters for a spacecraft that uses nuclear reactor as a means of propulsion (Bussard – Delauer, 1958):

$$P_c = \frac{g_c(\gamma - 1)}{2\gamma}(I_{sp})I_{tot}\,S\,\frac{\rho_{fuel}}{w_{fuel}} \tag{7.10}$$

This equation (7.10) can be thought as a performance analysis equation for a nuclear spacecraft. In this equation, there is the mixture density within the reactor core. This is the only nuclear coefficient that is found in the equation, but that variable ensures reactor criticality. Critical fuel density is essential for determining the cavity radius as well as the materials to be used in the reactor as can be seen in Fig. 7.6.

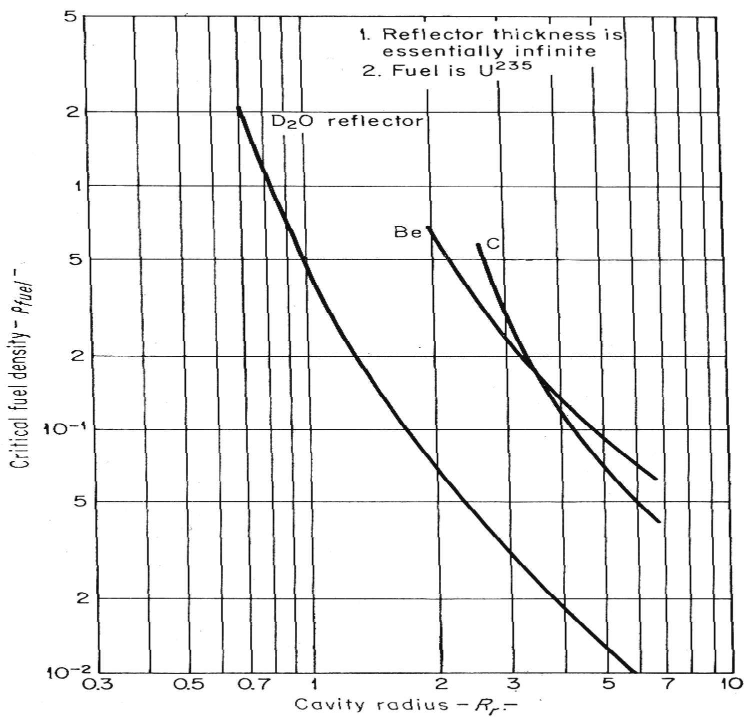


Figure 7.6 : Cavity Radius as Function of Critical Fuel Density

The equation must be analyzed further to get some meaningful results that can be interpreted. The below is a case example that can help to demonstrate the importance of the separation ratio for the reactor power density in the spacecraft:

g_c =9.81 m/sec^2 (This is the gravity acceleration coefficient at Earth)

γ = Characteristic specific heat ratio for Hydrogen = 5 / 3

ρ_{fuel} = 0.10 kg/m^3 (reasonable lower limit for a gaseous reactor)

Thus, the equation 7.10 with substituting the above values becomes:

$$P_c = \frac{9.81(1.6666-1)}{2(1.6666)}\left(I_{sp}\right)I_{tot}S\frac{\rho_{fuel}}{w_{fuel}}$$

By reordering the equation and by calculating the fixed coefficients:

$$P_c = 2.743\ I_{sp}\ I_{tot}\ \frac{S}{w_{fuel}}$$

This equation can be used for systems analysis. To further the example, it can be assume that there is an allowable nuclear fuel weight of W_{fuel} = 1000 kg. In addition, the need for a desirable specific impulse of I_{sp} = 6000 sec exists for the needed mission for the nuclear propulsion spacecraft. A total impulse of 2 x 10^7 kg sec can be approximated in view of the data published by NASA.

Thus, the equation above can be substituted with specific impulse and total impulse, so that it becomes:

$$P_c = 2.743(6000)(2x10^7)\frac{S}{1000}$$

The calculation of the above numbers results in:

$$P_c = 32.916x10^7 S$$

For Separation Ratio values of around 10^{-3}, there will be proper results for the chamber pressure. It is also possible to use this equation, other way around as a test separation ratio can be inputted to try to find the required W_{fuel} (weight of expelled nuclear fuel) from the spacecraft. This way, it is possible to calculate a cost function to see how much nuclear fuel is lost and its approximate cost for the mission requirements (Pike, 1998).

8. HEAT TRANSFER – FLUID DYNAMICS CONCEPTS IN GAS CORE REACTORS FOR PROPULSION IN SPACE

8.1 The Modes of Heat Transfer in the Reactor Matrix of the Spacecraft

One of the more important concepts in the design of the nuclear reactor matrix besides the separation ratio is definitely the concept of heat transfer. This is an important concept, as the whole idea of nuclear propulsion is actually based on the efficiency of heat transfer to the propellant fuel.

It has been already established in the preceding sections that the whole idea of nuclear propulsion was based on the principle that a nuclear reactor would be able to heat a propellant fluid (such as hydrogen) more efficiently. This means that the efficiency of the nuclear propulsion system is greatly dependent on the heat transfer of the nuclear reactor heat to the working propellant gas.

Thus, it is essential to examine the heat transfer processes in a nuclear propulsion system. This way, the performance criteria for the nuclear propulsion system can be established, as based on the heat transfer characteristics of the nuclear reactor in the spacecraft. In addition, the specifics of heat transfer must be understood to cite the performance criteria for the spacecraft.

There are mainly three different methods in which heat transfer can take place in the spacecraft. These three methods are conduction, convection, and thermal radiation. Sometimes, it is also possible for the heat to be transferred by thermochemical means too. For the purposes of simplicity, simplified versions of heat transfer will be used to get a general design parameter for the reactor chamber in the spacecraft.

The following can be assumed:

- Heat transfer by conduction takes place in a cylinder like environment, since the rocket design is based on a cylinder like design. Hence, it can be assumed that the reactor chamber to be like a cylinder for the intents and purposes of establishing a conduction heat transfer equation (Bussard, 1965).

- Convection is considered to occur in an ideal isentropic flow in order to have an ideal approximation. Convection takes place from reactor core to the gaseous propellant. In convection theory, gas to gas convection takes place by the kinetic theory of the gases. Kinetic theory of gases dictates that the speed of gases and their collisions will determine the amount of heat transfer that will take place (Bussard, 1965).

- Thermal radiation is also an important part of heat transfer. It is also possible to treat this as a simple black body problem to analyze the amount of heat that is radiated by thermal radiation. Especially in the instance of nuclear reactors, an important criterion is thermal radiation (Bussard, 1965)

The simplified initial conditions as well as boundary conditions must be inputted in accordance with the geometry of the nuclear reactor core and the geometry of the spacecraft to help form these equations. Then, the necessary integrations can be carried out and the resulting differential equations can be used to find approximate solutions. The final step is to make sure that the results from these heat transfer equations express the amount of heat imparted into the propellant fluid. This can help to realize the heat and temperature characteristics of the propellant fluid. This can also result in understanding the measure of the exhaust velocity that depends on the temperature of the fluid.

8.2 Conduction

Conduction is the main mode of heat transport in any physical system. Its main equation has been founded by Fourier. One of the advantages of using nuclear systems for propulsion is that symmetry will be required for the nucleonic operation of the reactor. That can allow ascertaining the heat transfer characteristics by simplifying the system to one or two dimensions (Bussard, 1965).

It can be assumed that the nuclear reactor is designed as generally a cylinder like physical object. The reason for this stems from the fact that most spacecrafts are oblong shaped and this causes the rocket design to be cylindrical in nature. Thus, it is possible to assume that the nuclear reactor is steady state with constant internal heat generation. However, the cylinder can be thought as infinitely long for simplification processes. The boundary conditions of the spacecraft can serve as the general limitations for conduction in nuclear propulsion (Thom, 1977).

8.3 Convection

Convection has to be examined too as a means of analyzing the amount of heat transferred to the propulsion fluid from the nuclear reactor in the spacecraft. It is essential to consider a boundary layer situation in which the heat transfer is completed by molecular conduction through a laminar layer on the surface (Thring, 1960).

It is possible to take the laminar flow of the propellant fluid and make some boundary layer simplification for design purposes. These boundary layer simplifications can include taking into account, a steady two dimensional compressible fluid flow with the properties defined as a function of time (Thring, 1960).

In order to achieve this, it is essential to use the Navier Stokes Equation of Momentum, the Continuity Equation, Energy Equation, and the State Functions. The derivation of these equations can be found in any advanced fluid mechanics book, as they are immensely important in defining the energy state of a fluid at any point n time in the flow (Thring, 1960).

There is also a serious interaction between the velocity and the temperature distribution in the propellant fluid flow. This is in concordance with the thermodynamic properties regarding gases, since velocity and temperature are carefully intertwined with each other. For the temperature to be up, the velocity must have a high value.

If the special condition of having a large velocity with a large Reynolds number and negligible buoyancy forces exists, then the flow of the propellant fluid is considered as forced and it can be assumed that there is a forced convection in the spacecraft combustion area. Free convection is the exact opposite, as it deals with low flow velocity, large temperature differences, and large buoyancy forces (Bussard, 1965).

In nuclear propulsion, there are very high velocities in the working fluid, which also happens to be the propellant fluid. Obviously, in a nuclear heat exchange flight propulsion system, there is a forced convection due to the facts below:

- There is high velocity on the working fluid (propellant gas) in order to create the necessary exhaust velocity and the corresponding momentum transfer in the spacecraft.
- The presence of temperature differences in the flow can be considered small, since the nuclear reactor will generate large and uniform heat for the propellant fluid in the spacecraft (Ellerbrock and Livingood, 1962).
- Due to the condition of the flow of the propellant fluid (such as H_2 hydrogen gas), the Reynolds number can be considered as very high (Ellerbrock and Livingood, 1962).
- There is some compression in the working fluid, due to differences in the pressure in the working chamber. Hence, the properties of compressible flow must be mentioned for the forced convection in the open chamber of the gaseous core reactor of the spacecraft (Ellerbrock and Livingood, 1962).
- There is also considerable friction that needs to be addressed in the flow of the propellant fluid, after it gets heated in the nuclear reactor.

By definition, in laminar flow, the fluid is moving only in the axial direction of the tube. In addition, in laminar flow, it can be assumed that the flow is taking place in a cylindrical environment with the velocity remaining constant on the cylinder surfaces. Hence, the flow and the corresponding pressure are axial to the surface in laminar flow (Ellerbrock and Livingood, 1962).

However, in turbulent flow, it is imperative to assume that the velocity components of the flow can become normal to the tube axis. Hence, this will cause a serious interaction between the axial and the normal forces within the fluid in a turbulent flow. But, much depends on the fluid's viscosity as well as other conditions like surface conditions (Thring, 1960).

It needs to be said that conditions of turbulent flow exist within the nuclear reactor of the spacecraft. Some conditions obviously benefit from this as the shape of the spacecraft and the corresponding nuclear reactor chamber can be conceived as cylindrical in nature (Bussard, 1965).

In addition, if the use of laminar heat flow exchangers are analyzed, it is possible to see that these types of heat exchangers will have a potentially large heat transfer capability and per unit pressure drop. Moreover, in laminar flow, it can be seen that the heat energy is transported from a solid surface into a moving fluid (which is consequentially the propellant fluid). This transportation of heat energy takes place by the means of the combined effects of both conduction and convection. Hence, it is also possible to use partial laminar flow to approximate the conduction and the convection of heat flow from the nuclear reactor in to the propellant. But, full turbulent flow solutions needs to be carried out in order to get specific design parameters for the spacecraft. (Bussard, 1965).

8.4 Thermal Radiation Heat Transfer

The third mode of heat transfer in nuclear propulsion systems in a spacecraft is definitely the presence of thermal radiation. Because of the fact that the temperatures are very high in rocket reactors, the presence of thermal radiation heat transfer becomes significant. Moreover, the ionization of propellant molecules along with the disassociation of fluid molecules can make thermochemical changes and heat transfer to be a force to be calculated (Grayamov, 1991)

In fact, if these effects are used properly during the design phase of the nuclear propulsion system, then the heat transfer to the propellant fluid from the reactor can be made more productive.

- The radiation at the propellant exit (which is obviously unshielded for design purposes) will cause an increased heat load in the downstream structure in addition to the heat transferred from both convection and conduction.
- The thermal radiation from the flow channel walls can be used to heat the propellant gas, so that higher temperatures can be reached in the flow.
- Due to the cylindrical geometry of the spacecraft, the radiation from the outer side boundaries of the nuclear reactor core will cause increased heat loads on the insulation barrier, as well as on the neutron reflector regions (Turchi, 1998).

- It is possible for the thermal radiation from the walls of the core system to increase the heat transfer to the propellant by more than % 20 depending upon the circumstances of the mission (Bussard, 1965).

8.4 Thermionic and Thermochemical Effects in Nuclear Propulsion

In rocket reactors, the system will have to operate at very high temperatures in order to guarantee that the necessary exhaust velocity will be sufficiently high enough for the required specific impulse. Hence, at such high temperatures, both thermochemical effects as well as thermionic effects take place. These thermochemical and thermionic effects include dissociation, ionization and electron emission. More importantly, all of these will have some effect on the heat transfer in the nuclear propulsion mechanism of the spacecraft (Bussard, 1965).

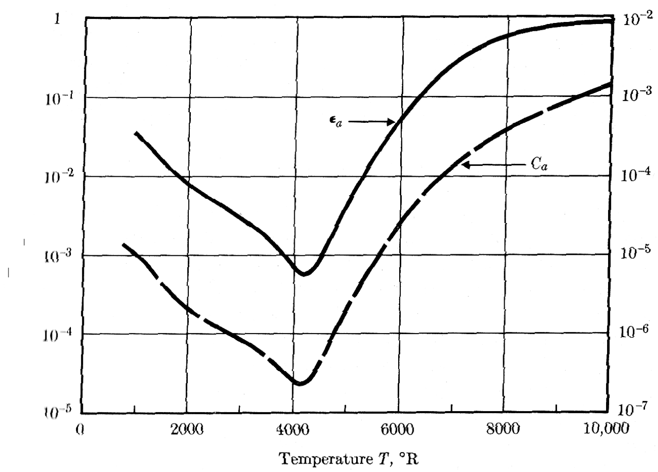


Figure 8.1 : Gas Absorption for Hydrogen under 100 atm Pressure (Bussard, 1965)

One of the most important effects as seen on the Figure 8.1 above, under these high temperature gradients is the dissociation of hydrogen. As it has been stated before, hydrogen is used as a propellant fuel in the spacecraft. The nuclear reactor heats up the hydrogen molecules in the propellant fuel and then they are then speeded up and ejected from the nozzle giving forward momentum to the spacecraft (Litchford, 2005).

Nevertheless, it is also important to note that there is diatomic hydrogen in any kind of hydrogen propellant. Hence, the diatomic molecules will be the perfect candidate for disassociation in the propellant fuel. Especially the inner layers of the exhaust chamber will be quite high and thus as a result, the diatomic hydrogen molecules that hit the surface will be broken into normal H atoms as a result of disassociation.

$$H_2 + 103 kcal / mol = H_2 \longrightarrow 2H$$

For each disassociation of H2, there will be 103 kcal of energy per mole that is released by the surface. Once these disassociated hydrogen atoms becomes free, they end up diffusing into the main velocity stream. Thus, they are also recombined at the main stream. Hence, once this recombination takes place, the 103 kcal/mole is released acting as a volumetric heat source in the fluid gas. In fact, this "volume heat source" can be quite significant (Bussard, 1965).

Besides the molecular conduction and the forced convection that takes place, there is also the thermal disassociation of the diatomic hydrogen. This appears as a new heat source, in addition to the heat generated by the nuclear reactor. In addition, as the wall temperature of the interior surface of the nozzle area becomes higher, the dissociation rate of hydrogen will be even greater than before (Litchford, 2005).

Hence, it is important to state that the disassociation of hydrogen due to high temperatures reached in the nuclear reactor will play a significant role in the performance of the spacecraft as it can be seen in the Figure 8.2 below.

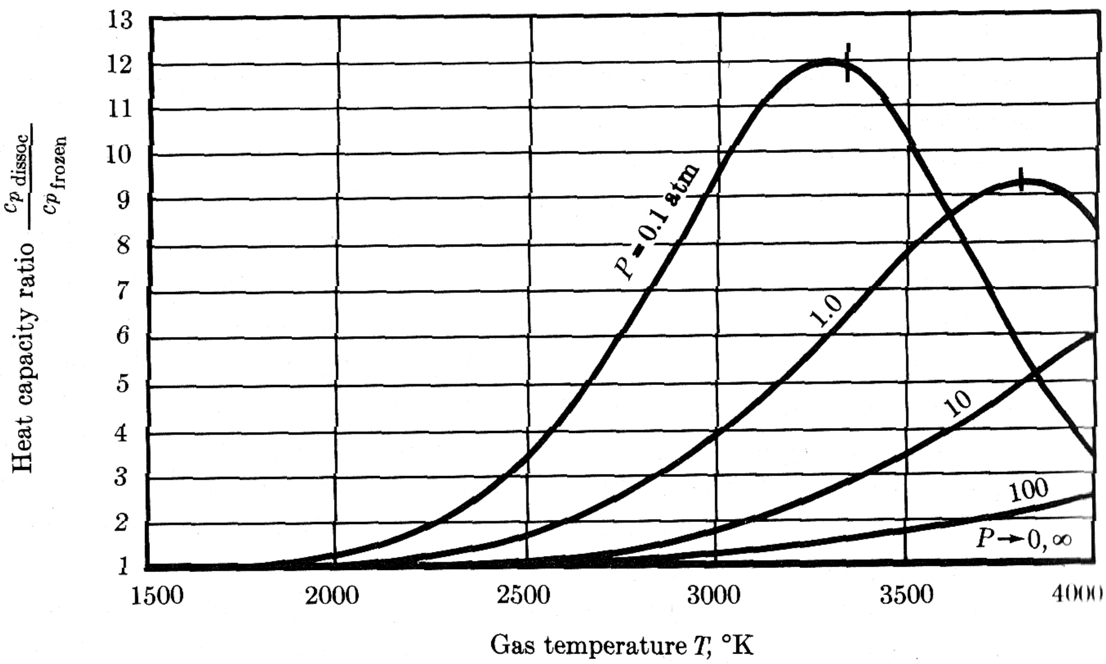


Figure 8.2 : Specific Heat Capacity of Disassociating Hydrogen (Bussard, 1965)

As a result, the atomic diffusion coefficient will also rise. Moreover, the increased heat transfer due to the disassociation – recombination effect will quickly become significant, and even small increases in the temperature can cause important volumetric heat to be produced by disassociation.

The temperature and the pressure of the fluid (propellant gas of hydrogen) will determine the equilibrium disassociation amount. For proper calculation of the heat coefficients, the hydrogen molecular collision rates will have to be analyzed along with the rate of the monatomic hydrogen diffusion. Secondly, the heat transfer equation will also have to be utilized for comparison with the non reacting flow systems.

However, another thermionic effect that needs to be discussed is definitely the surface ionization of molecules of the propellant gas H_2. In fact, at high temperatures that is experienced in a nuclear reactor, the ionization of hydrogen or H_2 can contribute to the overall energy transport in the propulsion system, more than the dissociation at lower scale temperatures (Litchford, 2005).

Thus, unless the good design of the reactor system gets destabilized, the phenomena of disassociation as well as ionization should be exploited in the heat transfer process for best results as seen in Fig. 8.3. Disassociation as described in the above section has more influence on the total heal transfer process after forced convection and conduction (Bussard, 1965).

As a final analysis, it is important to note that when a designer is calculating the heat transfer to the propellant in the nuclear reactor, it is essential to take convection, conduction, thermal radiation, disassociation, ionization, and absorption in to the account. FLUENT can be used to approximate the energy and heat transfer.

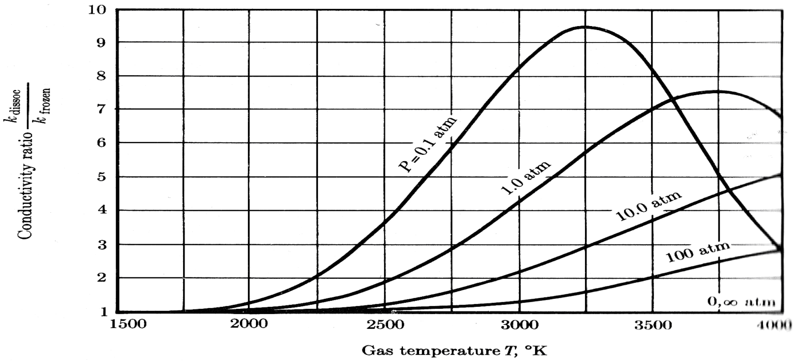


Figure 8.3 : Thermal Conductivity of Disassociating Hydrogen (Bussard, 1960)

8.6 Flow Characteristics and Viscosity

The following conditions can be assumed to affect the flow of the propellant in the spacecraft:

1) All modes of heat transfer including conduction, convection, thermal radiation, as well as ionization, thermionic processes and even the disassociation of hydrogen will need to be considered in the energy equations that describe the heat transfer within the nuclear propulsion system.
2) Due to the very high temperatures reached in the flow, viscosity of the propellant as well as the flow characteristics will be affected as seen in the Figure 8.4.
3) Very Large Reynolds number will be reached in the flow (Sutton, 1992).
4) Although, it's possible to use laminar flow modeling for simple fluid dynamics calculations, the type of flow that takes place is definitely turbulent flow. Hence, turbulent flow characteristics will need to be calculated in order to correctly approximate the flows within a nuclear propulsion system of the spacecraft.
5) Navier Stokes equations of fluid dynamics, as well as energy equation and equation of continuity will need to be solved for the analysis of momentum and energy aspects of the flow.

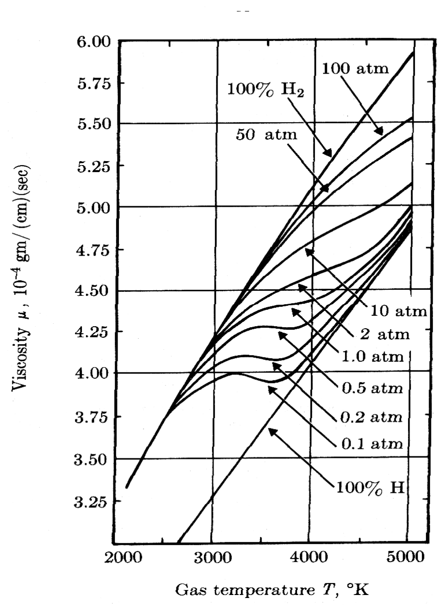


Figure 8.4: The Effect of Gas Temperature on the Fluid Viscosity (Bussard, 1965)

8.7 Heat Exchanger Design in Nuclear Propulsion Systems

In order to design a proper heat exchanger in the spacecraft nuclear systems, it is essential to ascertain information about the distribution of the propellant fluid (hydrogen fuel) temperature, wall surface and internal temperature, pressure of the flow from the reactor chamber, the speed of the flow, as well as the proper sizing of the heat exchange for the core.

The requirements of reactor neutronics, as well as the process of heat transfer are closely intertwined with each other. However, the problem of fluid flow reactor neutronics interaction is a difficult one, but it must be considered carefully in a nuclear propulsions systems design in a spacecraft. One of the criteria that can help the designer in their calculations is the reactor performance as related to heat transfer.

8.7.1 Mean Temperature Difference

During the design phase of a nuclear reactor core heat exchanger, the total heat transfer of the reactor is important. In order to achieve this, the basic convection equation of Newton needs to be integrated to turn into the differential equation:

$$dq = hdA\Delta T \tag{8.7}$$

In a constant fluid flow cross section, the heat transfer coefficient will depend on the physical properties of the propellant fluid and also upon the net total temperature. Thus, both enthalpy and the temperature difference is related to the value of q. Hence, by separating the geometric and thermal variables, (Bussard, 2006):

$$\frac{dq}{h \Delta T} = \int dA \tag{8.8}$$

dq : The heat generation distribution

ΔT : Variation of Temperature

h : Heat Transfer Coefficient

dA : Area distribution

However, the problem with this differential expression is the fact that there will be a nonlinear temperature dependence on h (heat transfer coefficient of convection) as well as the variation of ΔT . The non linear variation of the of the heat generation q will also be a factor that complicates the calculation of this function.

8.7.2 Geometric Considerations in the Heat Exchanger

It is essential to consider the weight as well as the mass of the nuclear reactor in the design of nuclear propulsion systems. Thus, as far as the nuclear reactor heat exchange design is concerned, this means that the compactness of the nuclear reactor core.

The geometric compactness of the nuclear reactor core is given by the equation:

$$\xi = \frac{A_h}{V_c} \tag{8.9}$$

The parameters for the equation are:

ξ: Compactness ratio of a nuclear reactor

A_h: Total heat transfer surface area

V_c: Nuclear Reactor Core bulk volume

It is possible to surmise from this algebraic equation, that as the ratio gets bigger, the heat transfer per unit volume or per unit mass will be greater too. Because of the fact that very high temperatures are reached in the nuclear reactor core, special design techniques will need to be used in order to achieve the highest possible compactness for the best possible heat transfer.

Thus, the designer will have to make sure that all possible geometries are explored for the best possible heat transfer configuration for the best possible compactness. In addition, compactness ratio is an effective tool in the hands of the space engineer. The design of a nuclear spacecraft is within not only a domain of the nuclear engineer, as it is also essential to reduce the total mass and the volume of the spacecraft as much as possible. Bigger mass and bigger volume will mean larger problems with shielding, propulsion, energy transfer, and logistics (Rom, 1962).

8.8 Power Density Distribution

In order to compare nuclear reactor cores, it is essential to use the ratio of heat transfer surface to total volume as a parameter of compactness. It is also possible to make a comparison by using the "*bulk core power density*" to quantify and compare nuclear cores with different geometries. Hence, power density equation can be expressed as:

$$K_c = \frac{q}{V_c} \tag{8.10}$$

The parameters can be defined as:

K_c = Bulk core power density

q = Thermal power output of the reactor core

V_c = Total nuclear reactor core volume

Hence, it can be stated that the reactor thermal power P can be expressed in Megawatts by the relation given by R.W. Bussard as:

$$q = C_q(1 - f_e) f_c P_r \tag{8.11}$$

where:

C_q = conversion factor

f_e = Fraction of the energy escaping through the reactor as photons and fast neutrons

f_c = fraction of the total thermal power released in the core

Hence, it is possible to write the equation for power core density:

$$K_c = C_q(1 - f_e) f_c \frac{P_r}{V_c} \tag{8.12}$$

where:

P_r = Reactor thermal power

In order to lay the groundwork for the heat transfer design problem, it is interesting to note the following equations to see the reactor power density, as well as the reactor power performance in the spacecraft.

For a reactor core that is operating at constant temperature T_r and with propellant gas entering at T_o and leaving the core at T_e; it is possible to write the bulk power density as given by Bussard in the following form (Bussard, 1965):

$$K_c = 2.93 \, x10^{-7} A_{sp} h_{cg} \Delta T_{LM} \quad \textbf{(8.13)}$$

A_{sp} = Specific Heat Transfer Surface Available Per Unit Core Volume

h_{cg} = Average heat transfer coefficient

ΔT_{LM} = Log mean temperature difference

It is now possible to express the log mean temperature with the following equation:

$$\Delta T_{LM} = \frac{(T_r - T_o) - (T_r - T_e)}{\ln \frac{T_r - T_o}{T_r - T_e}} \quad \textbf{(8.14)}$$

where:

T_r = The operating temperature of the core

T_o = Temperature of the propellant entering the core

T_e = Temperature of the propellant exiting the core

The average heat transfer coefficient for the propellant gas can be expressed by using the equation:

$$h_{cg} = \frac{\int_{T_o}^{T_e} h_c(T)\, dT}{T_e - T_o} \quad \textbf{(8.15)}$$

In order to calculate the bulk power core density, there are the separate equations for log mean temperature difference as well as for the bulk power density.

It is possible to write the weight of the core as a function of its volume and density:

$$m_{core} = \rho_c V_c = \frac{\pi}{4} D_c^2 L_c \rho_c \tag{8.16}$$

Where:

ρ_c = average bulk density at the core

D_c = Diameter of the core

L_c = length of the core

In general, if the preceding equations are combined in to Equation 8.11 :

$$P_r = K_c V_c \tag{8.17}$$

Then, the above equations are combined by placing K_c and V_c,:

$$P_r = 2.93x10^{-7} A_{sp} h_{cg} \Delta T_{LM} V_c \tag{8.18}$$

It is possible to make further substitutions for each variable to get power density for the reactor:

$$P_r = 2.93x10^{-7} A_{sp} \frac{\pi}{4} D_c^2 L_{c)} \frac{(T_r - T_o) - (T_r - T_e)}{\ln \frac{T_r - T_o}{T_r - T_e}} \frac{\int_{T_o}^{T_e} h_c(T) dT}{T_e - T_o} \tag{8.19}$$

This detailed equation for reactor power density is a function of the log mean temperature difference, as well as the core heat transfer coefficient.

Moreover, the presence of propellant fluids temperature entering and exiting the core, as well as the volume of the core geometry, play an important role in understanding the relation between the heat production and transfer within the nuclear reactor core and the propulsion mechanism of the spacecraft. This allows the possibility to construct a realistic picture of the inner workings of the nuclear propulsion system. It is feasible to use these equations to see how to design a system that gives the maximum possible temperature for the propellant gas with maximum possible dispersion in to the fluid (propellant) (Ellerbrock and Livingood, 1962).

9. GASESOUS CORE REACTOR FISSION NEUTRONICS IN PROPULSION

9.1 Advantages of Gaseous Core Reactors

One of the main advantages of gaseous core reactors is the fact that they can work n much higher temperatures as compared to solid core reactors or as compared to liquid core reactors. In addition, it has been proven that fission neutronics of gaseous core reactors is usually more superior to other conventional design such as solid core reactors (Norring, 2004).

It is also important to state that in space propulsion, the concept of microgravity becomes prominent as the reactor works in a weightless environment. Hence, the operation of a gaseous core reactor in a spacecraft is much more easier to control as compared to a regular type of a reactor. The neutronics as well as the thermodynamics of a gaseous core reactor can be performed more easily in a weightless environment. Moreover, the control dynamics are simpler and thus it requires less personnel and oversight to operate (Mohler, 1961).

9.2 The Selection Criteria for Gaseous Nuclear Fuel

One of the more important concepts in gaseous core reactors is the presence of using gaseous nuclear fuels. In the previous section, it was established that the best recommended fuel for use in these reactors is Uranium Hexafluoride or UF_6. However, for terrestrial applications, the usage of Uranium Tetrafluoride or UF4 is also recommended for control and neutronics purposes (Norring, 2004).

Uranium Hexafluoride is much more corrosive fuel type as compared to Uranium Tetrafluoride. However, Uranium Hexafluoride is a better agent in a microgravity environment as compared to Uranium Tetrafluoride. Moreover, the fuel density can be lower in Uranium Hexafluoride and yet the production of actinides is much higher as compared to Uranium Tetrafluoride. In addition, the requirements for lesser pressure are also important as UF_6 can work in lesser pressures as compared to UF_4.

However, there are several designs of gaseous reactor cores that operate with UF4 as compared to UF6. The designer will have to oversee both nuclear specifications as well as aerospace specifications for choosing the best fuel for the spacecraft. It is possible to see the comparison between two fuel types on the Figures 9.1 and 9.2.

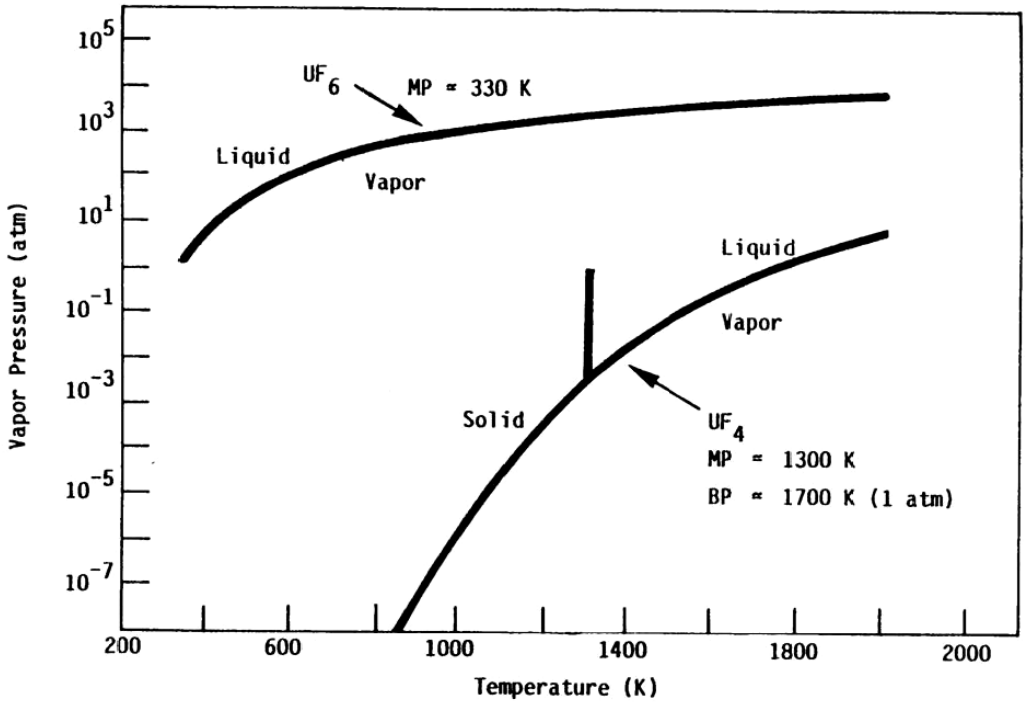


Figure 9.1 : Saturation Curves for UF4 and UF6

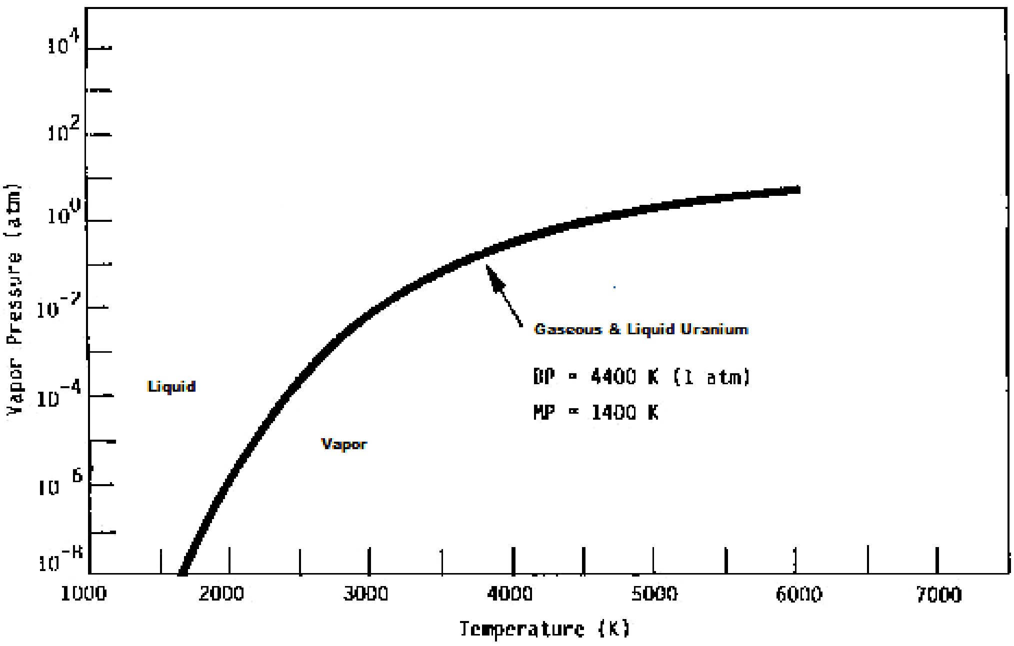


Figure 9.2 : Saturation Curves for Gaseous and Liquid Uranium

9.3 Considerations for UF6 Gas Core Reactor

In the design of a gaseous core reactor that works with Uranium Hexafluoride, it is essential to think about several important criteria. By altering these criteria, the design engineer will be able to control the neutronics and the reactivity of the gas core reactor, the power output of the reactor, as well as the temperature that is reached inside the core. These criteria include:

- The Geometry of the Reactor
- Pressure Inside of the Reactor Core
- Choice of Moderator and Reflector
- Enrichment of the Nuclear Fuel
- Fuel Feeding of the Reactor

9.4 Geometry Considerations for UF_6 Gas Core Reactor

The simplest way to affect the UF_6 gas core reactor is by changing its geometry. Through various researches, it has been shown that the best geometrical design is the usage of right cylinder shape geometry for best results. The best criticality can be achieved in this geometric design and more importantly, the neutronics control is much simpler with this design.

In geometry, the reactor dimensions are also an important consideration. In terrestrial applications, it has been shown that the best dimensions for the length of a right cylindrical geometry reactor core is between 3 to 5 meters. However, in space propulsion applications, it is essential to reduce the length as much as possible, as the length of the spacecraft will need to be limited for several purposes. Hence, the design of the geometry of the core should be in concordance with the design of the overall shape and the size of the spacecraft. Thus, the optimal design parameter for a UF6 Gaseous Core Reactor in a Spacecraft is definitely 3 meters with the shape of a right cylinder geometry (Norring, 2004).

9.5 Pressure Considerations for UF_6 Gas Core Reactor

The pressure inside the reactor can change from 70 atm to 200 atm. Generally, it has been seen that a UF_6 reactor will work with efficiency, if it is operated under these pressure regimes. However, contrary to popular belief, the rise in pressure does not constitute a rise in neutronics efficiency. In fact, under certain conditions, lower pressures can mean better neutronics efficiency and control.

This is quite important, as the maintenance of a high-pressure area inside of a spacecraft is quite a difficult feat inside of a spacecraft. In fact, it can introduce a lot of stability and dynamic control problems to the spacecraft. Hence, the ability to reach criticality and efficiency in neutronics of the reactor with low pressures can be essential in the design of the spacecraft (Mohler, 1961).

In fact, 70 atm is the recommended limit for pressure within the spacecraft as the integrity of the spacecraft structure can be effected less. Especially, the American spacecraft are specifically designed to work with lower pressures and thus a lower pressure in the UF_6 Gas Core Reactor will be better for the overall performance design criteria of the spacecraft (Duggins, 2007).

9.6 Reflector and Moderator Considerations for UF_6 Gas Core Reactor

Conventional thermal reactors usually use low mass number isotopes with small absorption cross sections relative to their scattering cross sections. The low mass number allows rapid loss of energy upon scattering, and the lower the ratio of absorption to total cross section allows for more neutrons to reach thermal energies without being absorbed in the moderator-reflector. Traditional materials include normal water (H_2O), deuterium (D_2O), beryllium (Be) and beryllium oxide (BeO).

The decision of material must be based on the specific design goals for the material. Faster spectrums have higher fission to capture cross section ratios, which lead to higher removal rates. The choice of BeO will allow the reactor to operate at a faster spectrum than traditional thermal reactors, while also allowing spectrum shifts if desired. The BeO can be processed into columns that can be externally adjusted to change the spectrum inside the core. By making the external moderator BeO instead of a liquid, there is no pumping required, which lessens the number of movable parts.

In addition, the usage of hydrogen as a propellant in the spacecraft, allows hydrogen to be used as a moderator due to its single proton mass. Hence, while BeO or other similar beryllium compounds are used as a reflector, the hydrogen propellant with its high thermal neutron absorption cross section can allow it to be used as a moderator in the reactor (Norring, 2004).

9.7 Criticality and Neutronics Considerations for UF_6 Gas Core Reactor

The core geometry consists of only two concentric right circular cylinders, containing the fuel and the reflector as shown in the Figure 9.3 below. The inner cylinder consists of the fissioning fuel (UF_6), and the outer consists of the reflector (BeO).

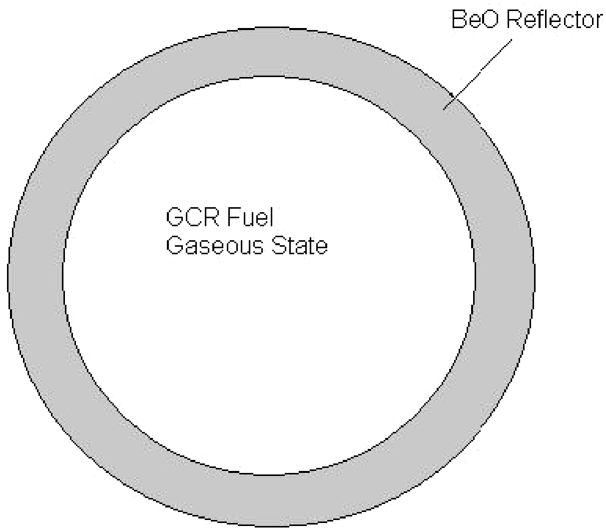


Figure 9.3 : Sample Geometry of a UF6 Gas Core Reactor

The UF_6 Gas Core Reactor design can be characterized as an externally moderated, circulating fuel reactor with a thermal neutron spectrum. There is no control or fuel rods or any other mechanical structures in a UF_6 Gas Core Reactor. There is simply a cylinder containing the gaseous fuel, and an outside cylinder containing the reflector material. Thus, the omission of fuel rods and control rods allows for a more simplistic design of the reactor in the spacecraft (Norring, 2004).

The system's criticality is a direct function of the core pressure. Therefore, any pressure leak will immediately cause a loss in reactivity. Controllers can directly change the neutronics in the core through a variety of methods. Rotating the BeO columns, changing the mass flow rate, and changing the reactor core pressure can be used to control core reactivity and neutron energy spectrum. These will allow the controllers to achieve optimum neutronics (Norring, 2004).

Reflector size and fuel density are optimized simultaneously, since varying one factor requires a change in the other factor in order to compensate for reactivity change. For instance, increasing the reflector size introduces a positive reactivity insertion and it must be offset with a decrease in fuel density for reactor control. Conversely, shrinking the reflector size requires an increase in fuel density to maintain criticality. It should be noted that instead of lowering the fuel density, the enrichment of the nuclear fuel could be adjusted. However, evaluating the system with three variables simultaneously significantly increases the difficulty in controlling and in approximating the system (Norring, 2004).

A smaller reflector will provide a faster neutron spectrum, because a neutron scattered into the reflector will either escape the reflector and thus disappear from the system or it will be reintroduced into the fuel region with a lower average number of scatters than when a larger reflector is present. The lower number of scatters is because of the fact that in a larger reflector, the neutron may backscatter towards the core from a point in space where if a smaller reflector were present, the neutron would have already escaped the geometry. The faster neutron spectrum also produces higher fission to capture ratios for the actinides, resulting in more fissions per interaction.

One benefit to a smaller reflector is an increase in plutonium generation, which increases the conversion ratio of the core (fissile material produced compared to fissile material depleted). A faster spectrum causes more U238 to undergo capture thus eventually creating Pu239, which has high relative neutron cross-sections within the energy regime that the UF_6 Gas Core Reactor operates. It should also be noted that instead of using static reflectors, using rotating BeO columns to increase or decrease neutron thermalization can control the spectrum of the core. However, controlling the nuclear fuel flow input is usually a more simpler way of controlling reactor criticality in a microgravity environment, since less number of mechanical parts are required.

A larger core (for a given gas pressure and enrichment) features lower leakage, as well as a faster neutron spectrum. The lower leakage is an obvious result of increased core size, since an increase of core size also increases the number of mean free paths in the core. This will cause more neutrons to interact before escaping into the reflector. The statistic probability of more neutrons interacting with the actinides will cause more fissions to take place in the reactor core (Rom and Ragsdale, 1962).

The faster spectrum is a result of more interactions taking place in the core without neutrons having seen the reflector region (without having seen the reflector region as often as would occur with a smaller core). Neutrons must pass through more mean free paths to reach the reflector and thermalize, increasing the chance that a fast reaction will occur. A larger core will raise k_{eff}, but the hardened neutron spectrum causes more actinide buildup in the core. The best reflector region is deemed between 10 cm to 50 cm for most efficient criticality. (Norring, 2004)

Table 9.1 : k_{eff} Values for 10 cm Reflected Core

Fuel *	Enrichment	U *	U235 *	U238 *	F *	P (bar)	K eff	%Rel Error	MFP (cm)
1.42E-04	10.00%	2.84E-05	2.84E-06	2.56E-05	1.14E-04	9.42	0.2133	1.41E-03	1210.5
1.42E-03	10.00%	2.84E-04	2.84E-05	2.56E-04	1.14E-03	94.16	0.4558	2.06E-03	131.2
1.42E-02	10.00%	2.84E-03	2.84E-04	2.56E-03	1.14E-02	941.6	0.7974	1.74E-03	12.899
3.40E-02	10.00%	6.80E-03	6.80E-04	6.12E-03	2.72E-02	2253.9	0.9048	1.21E-03	5.351
4.40E-02	10.00%	8.80E-03	8.80E-04	7.92E-03	3.52E-02	2916.81	0.9227	1.55E-03	4.129
8.40E-02	10.00%	1.68E-02	1.68E-03	1.51E-02	6.72E-02	5568.45	0.9553	1.25E-03	2.159
1.40E-01	10.00%	2.80E-02	2.80E-03	2.52E-02	1.12E-01	9280.74	0.9647	1.03E-03	1.295
2.80E-01	10.00%	5.60E-02	5.60E-03	5.04E-02	2.24E-01	1.86E+04	0.9668	9.90E-04	0.647
5.60E-01	10.00%	1.12E-01	1.12E-02	1.01E-01	4.48E-01	3.71E+04	0.9668	9.90E-04	0.324
1.20E+00	10.00%	2.40E-01	2.40E-02	2.16E-01	9.60E-01	7.95E+04	0.9668	9.90E-04	0.151
1.20E+01	10.00%	2.40E+00	2.40E-01	2.16E+00	9.60E+00	7.95E+05	0.9668	9.90E-04	0.015
1.20E+04	10.00%	2.40E+03	2.40E+02	2.16E+03	9.60E+03	7.95E+08	0.9668	9.90E-04	1.51E-05
1.42E-02	15.00%	2.84E-03	4.26E-04	2.41E-03	1.14E-02	941.6	0.9436	1.20E-03	12.923
1.93E-02	15.00%	3.86E-03	5.79E-04	3.28E-03	1.54E-02	1279.42	0.9996	1.26E-03	9.48
2.00E-02	15.00%	4.00E-03	6.00E-04	3.40E-03	1.60E-02	1325.82	1.0084	1.23E-03	9.145
2.50E-02	15.00%	5.00E-03	7.50E-04	4.25E-03	2.00E-02	1657.28	1.0422	1.36E-03	7.304
3.40E-02	15.00%	6.80E-03	1.02E-03	5.78E-03	2.72E-02	2253.9	1.084	1.48E-03	5.36

* Units of (atoms/barn*cm)	
Volume (m^3)	21.21
BeO Thickness(cm)	10

Mechanical properties favor a smaller core, as it is easier to manufacture, and to maintain pressure in compared to a larger core. The flow at which feed is introduced into the core affects both the criticality and the spectrum of the core. A higher flow rate will induce a positive reactivity insertion, as fissile fuel will be replaced faster after fissions. It is possible to see various interactions to k_{eff} with different core sizes in the Tables 9.1 and 9.2.

Table 9.2 :. k_{eff} Values for 50 cm Reflected Core

Fuel *	Enrichment	U *	U235 *	U238 *	F *	P (bar)	K eff	%Rel Error	MFP (cm)
1.42E-04	10.00%	2.84E-05	2.84E-06	2.56E-05	1.14E-04	9.42	0.986	2.12E-03	1010.8
1.45E-04	10.00%	2.90E-05	2.90E-06	2.61E-05	1.16E-04	9.61	0.9899	2.57E-03	992.09
1.50E-04	10.00%	3.00E-05	3.00E-06	2.70E-05	1.20E-04	9.94	0.9986	2.06E-03	962.65
1.52E-04	10.00%	3.04E-05	3.04E-06	2.74E-05	1.22E-04	10.08	1.0003	2.39E-03	908.47
1.55E-04	10.00%	3.10E-05	3.10E-06	2.79E-05	1.24E-04	10.28	1.0116	2.50E-03	934.71
1.60E-04	10.00%	3.20E-05	3.20E-06	2.88E-05	1.28E-04	10.61	1.018	1.96E-03	951.1
3.60E-04	5.00%	7.20E-05	3.60E-06	6.84E-05	2.88E-04	23.86	1.0207	2.06E-03	460.45

* Units of (atoms/barn*cm)

Volume (m^3)	21.21
BeO Thickness(cm)	50

Thus, it is important for the designer to oversee all of the possibilities mentioned above in order to control the criticality of the reactor. The neutronics of the reactor can be controlled through variety of factors ranging from the geometrical size and the shape of the reactor, to the configuration of the reflector as well as the moderator. A summary of criticality conditions for a UF_6 Gaseous Core Reactor in microgravity environment can be presented as:

- The best geometric shape for reactor neutronics in UF6 gas core reactor in microgravity conditions is a right cylinder.
- The length of the cylinder of the reactor cores is assumed to be 3 meters in a spacecraft for optimal design conditions.
- The fuel is gaseous Uranium Hexafluoride due to its more compatible properties in a microgravity environment of a spacecraft.
- The moderator is the hydrogen in the propellant fuel, while the reflector is a Beryllium compound such as BeO.
- There are no control rods or fuel rods in a UF_6 gas core reactor, since these are not needed to control neutronics and criticality. Moreover, the operation of control rods can be cumbersome and technically difficult in a spacecraft operating in a microgravity environment.
- The criticality of the reactor is controlled through the pressure in the core and the amount of fissile material that is pumped in to the core region. The introduction of fissile material will increase the criticality of the reactor since more fissile material will be available for fission. However, the separation factor mentioned in the previous section will also need to be calculated in

concordance with these criteria. Although, the presence of fissile actinides will increase the criticality of the reactor, it will also increase the amount of fissionable material that is ejected from the core through the nozzle along with the propellant. This is not desired, as the increase in the amount of fissionable materials that is ejected before fission will increase the cost of the operation by hundreds of thousands of dollars. Thus, it is essential to achieve maximum criticality without increasing fuel density.

- The nuclear fuel can be enriched to increase criticality, although this will also increase the cost of operating a UF_6 reactor in a spacecraft. Using low enriched fuel is the safest from a proliferation aspect, and cheapest to produce; however, it may produce a problem in pressure control. Because more U238 is introduced into the core than depleted, the pressure of the core will gradually rise. Moreover, if the enrichment of the feed fuel is lowered, then the rise of pressure becomes more dramatic, and eventually this will strain the mechanical limits of the core. (Rom and Ragsdale, 1962)
- Reactor must remain critical at all step points (it should actually be supercritical at the start of every step to account for burn up between steps). K_{eff} for the core should remain between 1.0 and 1.05, with an error less than 1%. (Norring, 2004)
- Reflector size and fuel density are optimized simultaneously.
- Loss of pressure in the reactor will immediately cause the system to abort since the loss of pressure will immediately decrease the criticality of the reactor.
- A low-pressure core exhibits a more thermal spectrum of neutrons and from a spacecraft point of view, it is much more easier to design the spacecraft with a lower pressure core.
- The more thermal spectrum results in less leakage from the reflector, thus lowering radiation damage to any containment vessels and the surrounding environment. This is also desirable for the shielding point of view in the spacecraft since any extra shielding will increase the design complications in the spacecraft.

10. NUCLEAR SHIELDING FOR NUCLEAR PROPULSION SYSTEMS

10.1 The Importance of Radiation Shielding in Nuclear Propulsion

One of the last important concepts that need to be mentioned in this thesis is definitely the usage of shields. When there is a nuclear reactor onboard a spacecraft, it is essential to shield the excess radiation and heat from reaching the payload and crew compartments. Especially in view of the fact that nuclear spaceships will be primarily used with a human crew, it is imperative to take into safety considerations of humans, as well as the safety and the durability considerations of the materials used in the spacecraft.

The human crew aboard a nuclear spacecraft will be subjected to both the cosmic radiation of space, as well as to the radiation that is generated in the nuclear reactor. Thus, it is essential to take in the mission duration requirements to design a proper shielding for the spacecraft. Hence, a mission that takes 3 months to complete will need better shielding, as compared to a mission that will take only one week to complete (Czysz, 2006).

It is also important to consider the fact that the shielding will take mass and volume from the total structure of the spacecraft. Since the designer will also have to allow for enough space to support the crew, the payload as well as the propellant, there may not be enough volumetric space for the shielding. Hence, the designer will have to walk a fine line trying to balance the needs for safety, with the need to preserve space for the overall mission of the spacecraft. Don’t forget that a spacecraft will cost billions of dollars and thus every little detail can be important for the design of the spacecraft (Taylor, 2009).

In some cases, direct shielding of the reactor core will be made possible, while some designs will incorporate indirect shielding techniques. The simplest flight shield design problem centers on protecting the crew of the spacecraft from the vacuum of space, the radiation from the nuclear reactor and from the cosmic radiation emanating from interstellar space. The shielding must incorporate the whole spacecraft to protect the astronauts per the Figure 10.1, but certain portions such as the exit area of the nozzle must also be shielded with extra care.

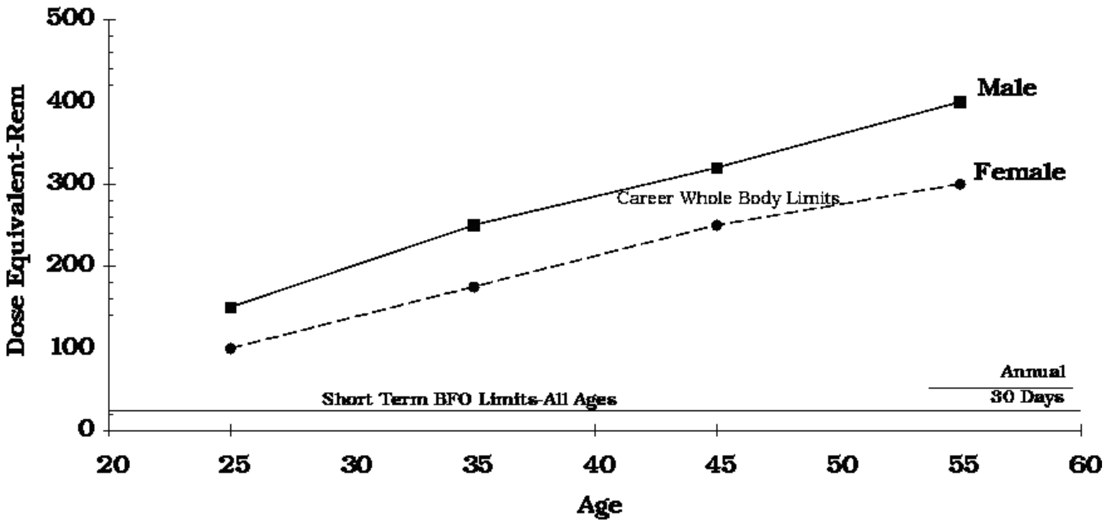


Figure 10.1: Recommended Radiation Dose Limits for Astronauts

10.2 The General Principles of Reactor Shielding

In a typical fission reaction, many high-energy particles with and without mass are ejected into the outer environment. These particles include various fission fragments such as neutrons, electrons, alpha beams, beta rays, photons, X rays, and Gamma Rays. In fact, it is possible to think of the reactor as a (p) point source that is radiating in an isotropic manner (Bussard, 1965).

Thus, the intensity of the radiation, which is defined as particle fluxes that is the number of particles emitted per unit area and unit time will reduce as the square of the distance

$$\text{The Effect of the Radiation flux} = \frac{1}{d^2} \qquad \textbf{(10.1)}$$

In the equation above, d is the distance from the reactor.

In most cases, especially in spacecraft, the d value may be impracticably large and this can cause certain problems. Hence, it is imperative to use strong shielding to protect the crew from being affected from fission fragments emitted as radiation.

In a typical fission reaction, there will be the primary fission fragments, as well as the radioactive decay of these fission fragments. Hence, the effects of the radiation and its shielding problem can be quite complex. A well-designed shield should slow down fast neutrons enough, so that they can be captured by the shield nuclei. Moreover, energy of the gamma photons should be absorbed as well since gamma rays are the most dangerous as seen in Figure 10.2.

Both the gamma rays and the neutrons are the most dangerous part of the radiation that is emitted and any shield that is designed to stop them can also stop alpha and beta radiation too. Thus, designing the ship's shield to protect against the dangerous gamma rays will also help to have a shield that will protect against other dangerous radiation as well.

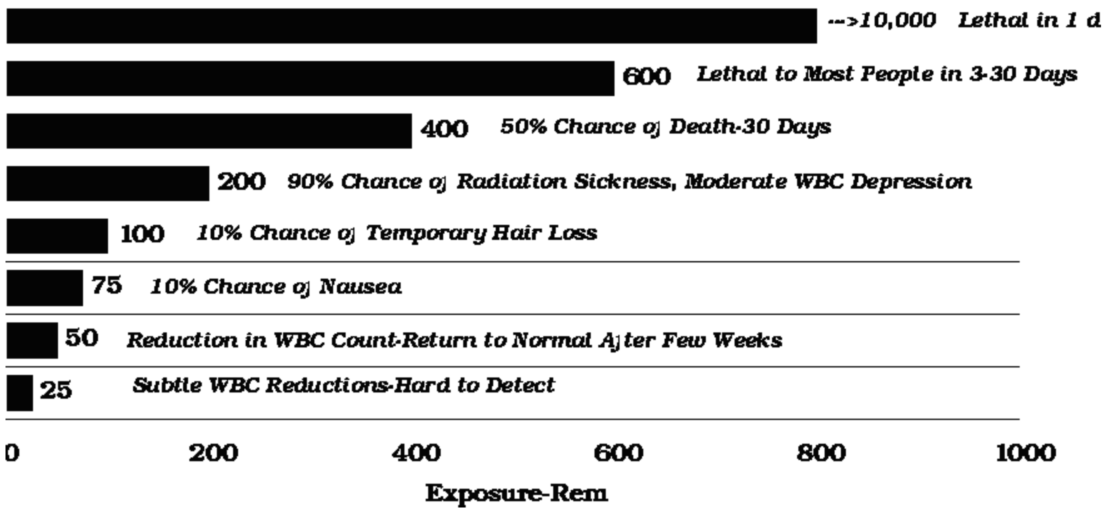


Figure 10.2 : The Effects of Gamma Radiation in Humans

In a spacecraft, the shield surrounding the nuclear reactor does not necessarily have to be separate from the nuclear reactor used in the space propulsion. In fact, the shield may include the propellant as well as the propellant tank.

Hence, it is possible to see a diverse representation of various fission fragments that are produced in a fission reaction in a spacecraft. Therefore, as it can be seen, there are various effects that need to be countered in shielding applications in order to create the best possible shield to protect the human crew that may be traveling onboard.

The shielding materials that are used must be chosen with care, so that total protection against all of the radiation types depicted below in Fig. 10.3, can be implemented (ranging from high energy gamma particles to various sub atomic particles coming from the cosmos). In addition, the materials that are used must be able to withstand these different radiation stresses for the duration of the mission along with the other safety parameters. However, the selection of a multi usable shield will be foremost in the design consideration of the nuclear spacecraft (Czysz, 2006).

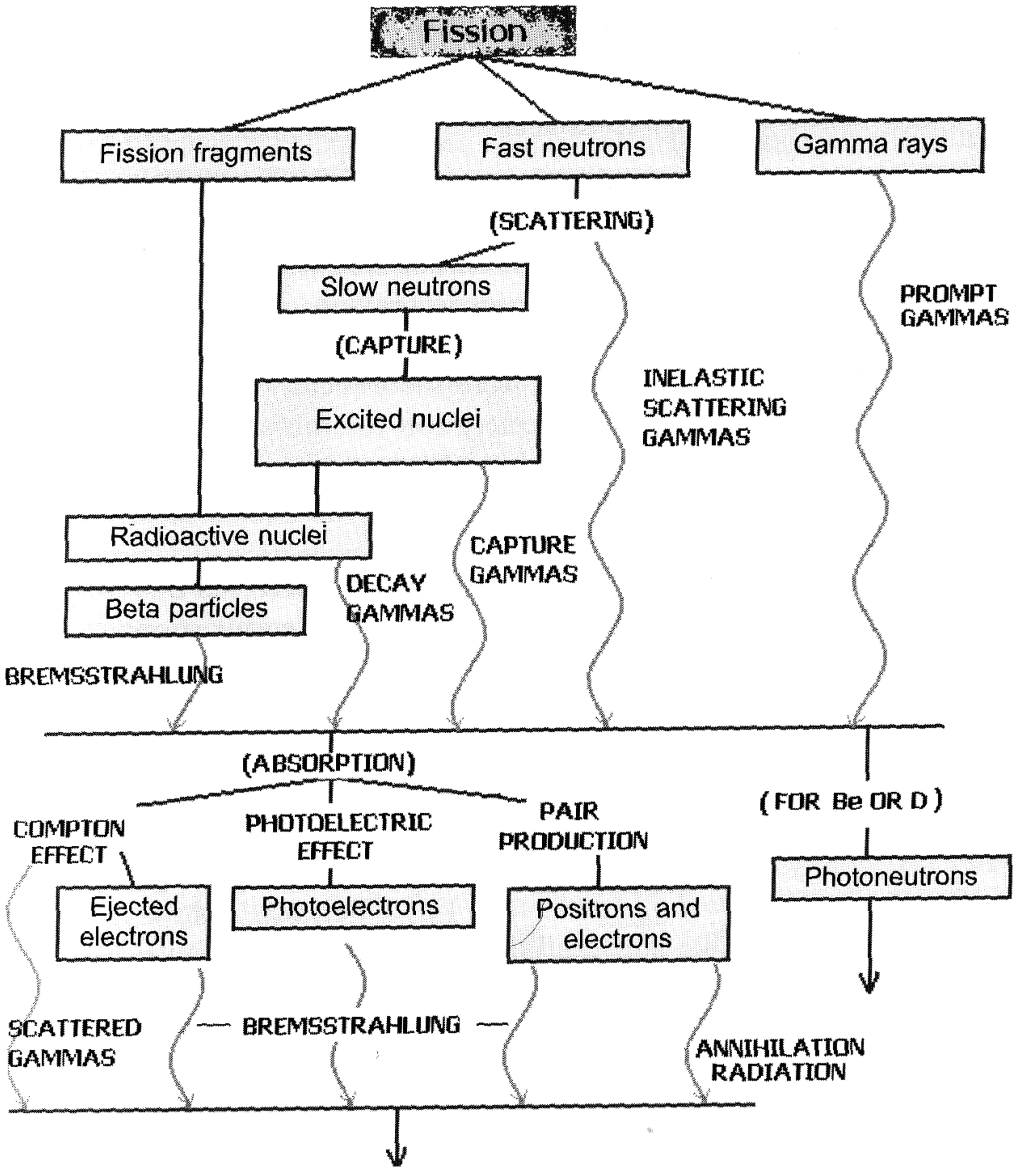


Figure 10.3: The Types of Radiation Emitted From a Fission Reactor (Turchi, 1998)

In general, it is possible to describe absorption in the spacecraft shielding with the differential equation:

$$dI(x) = \mu I(x)\, dx \qquad \textbf{(10.2)}$$

I(x) : local intensity of the radiation

μ = Line Absorption coefficient

dx = distance crossed

dI = change in the intensity of the radiation

In a nuclear system, neutrons are the hardest materials to stop. The reason for this is because neutrons are not electrically charged and thus they interact little with matter. The interaction of neutrons with the nuclei is ruled by the collision cross section or Θ. This cross section value depends on the energy and also on the type of the nucleus.

In general, the equation for neutron absorption in shields is given with the basic expression:

$$dc = I(N\, dx)\, \Theta \qquad \textbf{(10.3)}$$

I = neutron flux

N = volumetric density of the nuclei

N dx = area density of the nuclei

Θ = absorption constant

For example, it is possible to consider using a shadow shield by putting the reactor and the crew compartment at opposite ends as seen in the Fig. 10.4 below.

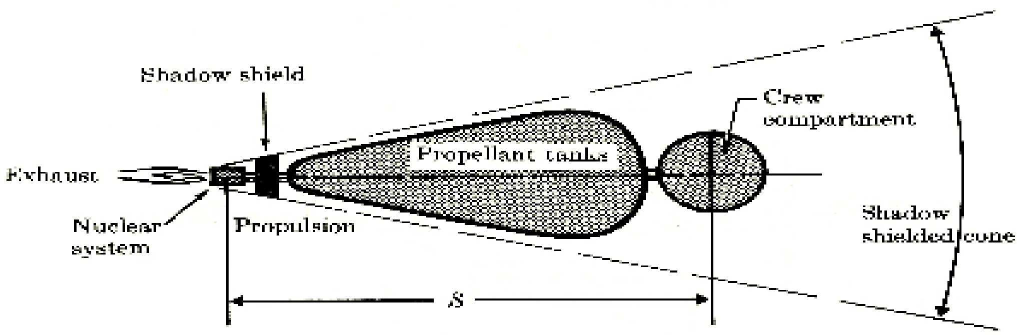


Figure 10.4: Using a Shadow Shield for the Nuclear Reactor in the Spacecraft

Thus, the shielding problems involve complex neutron and nuclear physics and thus it can be more difficult. Moreover, in a spacecraft, the technical problems are even more compounded since it is essential to limit the amount of mass in the spacecraft, so that the necessary payload can be included. Hence, as compared to terrestrial applications, the shielding problems for the spacecraft using nuclear propulsion are more numerous.

11. MATERIALS TECHNOLOGY FOR NUCLEAR PROPULSION

The usage of proper materials technology for nuclear propulsion is important. This stems from the fact that the limitations of the performance of nuclear propulsion systems are totally dependent on the structural design parameters of the materials used in the nuclear propulsion systems. Hence, the future of nuclear propulsions systems for spacecraft also lies in the production of highly tensile, temperature and corrosion resistant materials. As the strength of the materials increases, so does the operating efficiency of the nuclear propulsion systems. In addition, it is important to consider the fact that the material must be designed to be resistant to the vacuum of space as well as the cosmic radiation that can be found in the microgravity environment.

The proper usage of nuclear propulsion materials is subdivided in to four categories:

- The Materials Technology for Fuel Elements
- The Material Selection for Moderators and Reflectors
- The Materials Technology for the Structure of the Spacecraft and the Nuclear Components
- Neutronic Control Materials

11.1 Materials for the Fuel Elements

The performance of the nuclear reactor in the spacecraft is dependant upon reaching high temperatures. However, this also means that the fuel elements must be able to heat the coolant and the propellant gas to high temperatures. It is essential to note that there are shear force components in the viscous fluid flow in the spacecraft and thus the fuel elements has to be able to withstand these loads caused by the pressure differences in the coolant flow. In addition, the fuel elements will be subjected to internal temperature gradients and this can also cause thermal stress (Bussard, 1965)

In fact, it can be stated with certainty that at the high power densities that are required in the spacecraft, there will always be extreme values of temperature gradients, extreme core heat removal rates, and other thermal stress phenomenon.

In addition, in a nuclear rocket engine, it can be essential to power up the nuclear reactor to full capacity and to maximum temperature in the order of few tens of seconds. In fact, this procedure may have to be done continuously and in order for the fuel element materials to withstand these thermal stresses; the fuel elements have to relieve these thermal stresses by creep (Czysz, 2005).

The best definition of the concept of fuel elements would be that they contain the necessary fissionable material that supplies the power to the reactor when it is operated at the critical nuclear condition. Thus, the fuel elements must not compete with fissionable material for neutrons, so that the requirements for reactor criticality do not change. The base material of the fuel elements must be able to aid in the slowing down of the neutrons, so that the size and the weight of the reactor system can be reduced to proper and acceptable dimensions (Bussard, 1965).

Also, another important consideration is the fact that fuel elements must also be able to withstand the effects of the propellant such as hydrogen. Unfortunately, hydrogen is a very volatile material that has the capacity to embrittle materials, hydride surfaces and forms volatile compounds. Thus, a successful spacecraft reactor fuel element must be able to withstand the negative effects of the high temperature hydrogen propellant, without any structural damage or without any significant loss of fissionable fuel (Bogard and Lanz, 1962).

In short, it is possible to state the parameters of a good fuel element as:

- A fuel element must have high strength at high temperatures, as well as at low temperatures.
- The fuel element should have high conductivity in order to decrease the internal temperature gradients.
- The fuel element should be a poor absorber of neutrons.
- The fuel element must be a good neutron scatterer
- The fuel element must be able to withstand the corrosive effects of the propellants such as hydrogen

- The fuel element should not lose the characteristics described above with the addition of fissionable materials.
- The fuel element must be able to contain fissionable fuel

It is important to note that the operation parameters at high temperatures limits the use of reactor fuel elements severely in spacecraft. Some materials that can be used in fuel elements include Graphite, Tungsten, Tantalum, Molybdenum, Niobium, and Rhenium. Especially, the metallic carbides all appear to have a good thermal shock resistance characteristics and they are relatively stable at high temperatures in a high speed hydrogen flow. There is also some advanced work to suggest that the usage of Carbon 60 and Fullerenes can provide better fuel element characteristics, but the research concerning this is severely limited at this time. However, if a totally gas core reactor is used, there wont be any fuel elements as only gaseous form of nuclear fuel will be used without fuel rods or control rods.

11.2 Selection of Materials for the Moderators and the Reflectors

The energy of fission neutrons is lost to the moderating material by collision with the nuclei of the moderator. Moreover, gamma ray absorption in the reactor deposits energy throughout the volume of the moderator. Just like the fuel elements, internal temperature gradients will increase and give rise to thermal stresses in the moderator material. In addition, a moderator has to be chemically compatible with the propellant in the spacecraft, since propellant also acts as a coolant for the removal of the neutron and gamma heat energy (Bussard, 1965).

Thus, some characteristics of some good moderators include:

- Moderator – reflector material must have low atomic weight.
- Moderator – reflector must have small neutron absorption cross section
- Moderator – reflector material must have high conductivity
- It must be compatible with the propellant used in the spacecraft

One of the important characteristics of the hydrogen, which is used as a propellant in the spacecraft, is that it can also be used as a moderator. Because of its single proton mass, hydrogen can act as an excellent moderator. However, because of the high forward scattering taking place in the neutron proton collisions, hydrogen cannot act efficiently as a good neutron reflector. It is also significant to state that hydrogen also has a high thermal neutron absorption cross section and it makes it suitable for use in sections thick enough to thermalize the neutrons, but not enough to absorb the neutrons in the nuclear reaction.

11.3 Materials for the Core and Spacecraft Structure

There are two main structural components to a nuclear reactor working in a spacecraft. These are mainly the external pressure shell and the internal core support structure. The energy from the gamma absorption is generally deposited directly into the pressure shell. It can be stated that the strength and the density ratio of the shell material should be as large as possible, so that the pressure shells can have minimum mass and weight configuration.

Especially the point of nozzle attachment to the external pressure shell can be a point of local high stressing, due to the high thermal stresses caused by the gamma heating in these regions (where it is thicker than normal). In addition, local points of high loads can occur in the external pressure shell.

Another important design parameter is the internal core support structure. The internal core support structure is always immersed in a corrosive propellant gas such as Hydrogen. In addition, the internal core support structure must be able to carry the pressure drop and the differential expansion loads of the nuclear reactor core. It is important to cool the internal core support structure, in order to remove the internally deposited gamma heat energy. The support structure must be able to withstand the loads imposed on it, so that the fuel element design geometry is not effected in any significant way.

Unfortunately, total core loads can be extremely high; but since the core support structure is not involved in the nuclear reaction, the materials can be chosen more easily since nuclear considerations don’t come into play. The materials used in the core support structure must have high strength / density ratios and the materials must exhibit strong thermal conductivity. In addition, the core support structure must be able to withstand the corrosive properties of the propellant used in the spacecraft.

It is essential to note that the thermal stresses as well as the volumetric heating and the heat removal are the primary consideration in the choice of the structural materials. The internal temperature difference is proportional to the volumetric heat generation rate and the square of some characteristic thickness of the component divided by the thermal conductivity of the material. Especially for the pressure shell and for the structural component materials, the thickness is inversely proportional to the material yield strength. The only nuclear characteristics for the structural materials are the gamma absorption coefficient and the neutron capture cross section. Some physical properties of structural materials that can be used is seen in Table 11.1.

Table 11.1 : Physical Properties of Some Structural Materials (Bussard, 1965)

Material	Melting point T_m, °R	Sp gr at room temp, ρ/ρ_{water}	Short-time tensile strength, lb/in.^{2}		Yield strength, lb/in.^{2}		Stress for 1% creep in 10 hr, lb/in.^{2}		Young's modulus, 10^6 lb/in.^{2}		Mean linear-expansion coef, 10^{-6} °F^{-1}		Thermal conductivity, Btu/(hr)(ft)(°F)		Poisson's ratio ν	Thermal-stress parameter for gamma-heated structures†	
			σ_a	°R	σ_y	°R	σ_c	°R	E	°R	α	°R	k	°R		$\bar{\sigma}_2$	°R
Aluminum,	1650	2.71	13,000	500	5,000	500	3,000–	850	10	500	11	500	128	500	0.33	175	500
280			7,000	850	3,000	850	4,000‡	....	8	1000	14	1000	122	1000		2,700	1000
			3,000	1050	1,800	1050											
Aluminum,	1400	2.79	60,000	500	50,000	500	18,000‡	850	10.5	500	11	500	70	500	0.33	0.34	500
17S-T4			24,000	850	17,500	850	....	....	8.4	1000	14	1000	84	1000		100	1000
			7,500	1050	5,000	1050											
Stainless steel,	2950	7.92	76,000	500	31,000	500	24,000	1500	29	500	8.9	650	8.5	650	0.29	72	500
type 316			71,000	1350	22,000	1350	3,000	2000	24.5	1350	9.6	1550	12.1	1350		95	1000
			25,000	2000	17,000	2000			21.3	2000	10.9	2250					
Stainless steel,	3100	7.70	67,000	500	35,000	500	7,000	1500	29	500	5.6	650	13.8	650	0.29	18	500
type 430			48,500	1350	28,500	1350	800	2000	25	1350	5.8	1550	14.7	1350		29	1000
			6,000	2000	2,000–	2000	....	....	18	2000	6.7	2250					
					3,000												
Titanium,	3350	4.64	152,000	500	140,000	500	50,000–	850	16	500	4.7–5	500	8.2–9.9	500	§	0.14	500
Ti-150A			93,000	1000	69,000	1000	70,000	....	....	1500						0.94	1000
			40,000	1500	26,000	1500	10,000–	1250	11								
							20,000										
Inconel X.....	2900–	8.3	164,000	500	102,000	500	35,000	1800	....	....	7.5	500	8.3–9.4	500	§	1.8	500
	3050		136,000	1500	84,000	1500	13,000	2000	....	....	9.2	2000	20	2000		1.7	1000
			64,000	2000	54,000	2000											
Inconel.......	3000	8.5	90,000	500	36,500	500	57,000	1250	31	500	6.4	650	8.7–9.4	500	§	34	500
			80,000	1500	27,500	1500	19,000	1500	25	1500	....	....	13	2000		47	1000
			18,000	2000	11,000	2000	1,000–	2000	15	2000							
							2,000										

11.4 Materials for the Neutronic Controls

In order for any type of nuclear reactor to operate efficiently, it is essential for it to have some means of controlling its reactivity so that safe start up, shutdown and the safe operation of the reactor an take place. This is accomplished by neutron poisons that have a large neutron capture cross section, so that the high neutron flux and the reactivity of the reactor can be controlled. However, any neutronic control that has a high neutron capture cross section will also become a volumetric heat source due to the neutron absorption (Bussard, 1965).

This volumetric heat can cause significant thermal stresses in the neutronic control material. In addition, gamma radiation from the fissions taking place in the reactor core can be absorbed by the neutronic control material in direct proportion to the material density. In addition, the neutronics materials must be insensitive to neutron temperature so that the best possible control can be maintained.

11.5 Spacecraft Material Considerations

The materials requirement for the spacecraft is closely linked with the materials requirement for the nuclear components described above. The whole structure of the spacecraft (especially the structure that is nearest the nuclear core and near the nozzle exit area) will be severely affected by the nuclear reactions that are taking place.

As outlined above, there will be high thermal stresses caused by the high internal temperature gradients. In addition, the disassociation of the hydrogen propellant can also introduce some other corrosive and volatile effects and stresses in to the spacecraft structure. The liquid propellant is vaporized and then it is reduced to low density, as soon as it reaches the reactor core through radiation heating. In fact, when the propellant travels few centimeters in to the reactor core, it will immediately become gaseous. The spacecraft structure must take these corrosive effects of gaseous hydrogen in to consideration (Ellerbrock and Livingood, 1962).

Moreover, the superheated hydrogen that comes out of the reactor core will exhibit properties such as corrosion and erosion. In fact, superheated hydrogen will react with any exposed carbon surfaces and it will immediately form hydrocarbons. Thus, the use of graphite or fullerenes in the spacecraft must be considered carefully.

Another important effect of the nuclear reactor on the overall materials used is the fact that a nuclear core will cause much higher thermal stresses in the overall structure of the spacecraft. These thermal stresses will be much more higher as compared to chemical propulsion and there will be even a higher stress in the nozzle exit area. There will be serious deformations and strains placed on the spacecraft material and there will also be forced distortion (Ellerbrock and Livingood, 1962).

Thus, thermal stresses can cause creep or the plastic flow of the material within the spacecraft. But, although creeping can decrease thermal stresses, it can also introduce new vectored stresses caused by the dynamics of the spacecraft and it should be avoided as much as possible. Hence, the balance must be maintained between the creep rate to protect against thermal stress, total strain on the material, elasticity factors of the material and instantaneous stress at anytime *t*. Carbon 60 and some new titanium composite alloys have been considered in the overall superstructure of the spacecraft so that the necessary material requirements can be met for nuclear propulsion in a microgravity environment (Czysz, 2005).

In addition, the overall structure of the spacecraft must also be able to withstand radiation damage. No amount of radiation shielding can protect the super structure of the spacecraft completely and thus the material used in the outer body must be able to withstand radiation damage. Moreover, the materials used must be able to withstand the effects of the cosmic radiation outside of the spacecraft as well as the vacuum coldness and the pressure drop of space.

However, it is a known fact that materials will become more brittle with continued exposure to the radiation and the physical and the thermal properties of materials may also change. There could even be size changes in the material. NASA is considering Titanium and Fullerene alloys for best protection against radiation damage on the overall structure of the spacecraft.

12. SAMPLE DESIGN OF A NUCLEAR ROCKET SYSTEM

12.1 Aerospace Design Parameters for Nuclear Propulsion in Spacecraft

Now, after spending time for laying out the groundwork for neutronics, heat transfer, thermodynamics, and kinetics for the nuclear reactor in the spacecraft; it is possible to lay the optimum design parameters for the purposes of this thesis. The ideal conditions are used in order to approximate some of the solutions laid out on the above sections.

The sample nuclear rocket can be envisioned as a cylindrical shape rocket that is shown in Figure 12.1.

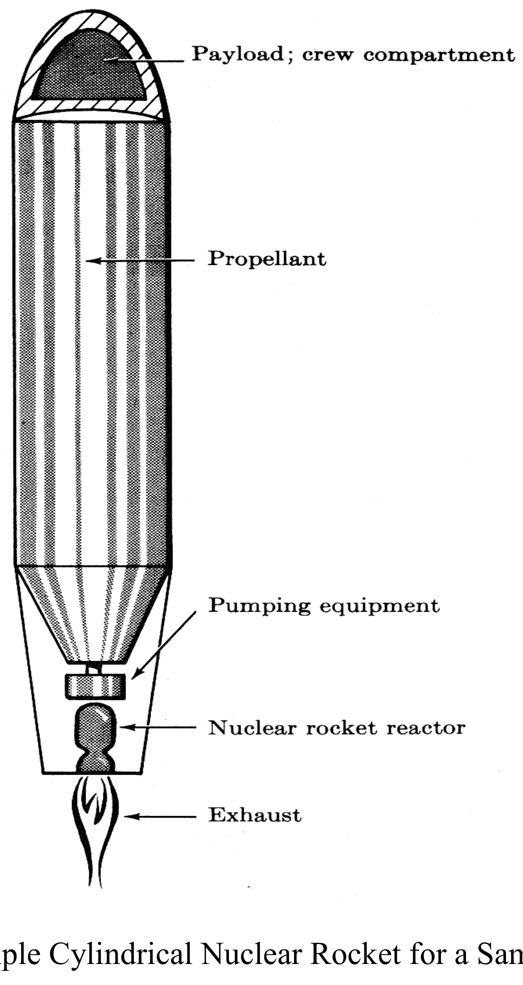


Figure 12.1 Simple Cylindrical Nuclear Rocket for a Sample Design

By proper approximation, it is possible to get a feel on concepts such as:

- The amount of the heat generated in the nuclear reactor core

- The heat that is transferred to the propellant

- The general heat distribution in the core and in the exit chamber through conduction, molecular convection and from ionization and thermal radiation.

- The amount of nuclear fuel that is fissioned and the amount of fuel that is not separable from the propellant during the ejection process

- General thermodynamic conditions in the exit nozzle and in the exit chamber of the nuclear reactor core

- The general thermodynamic parameters of the propellant gas

- The temperature and the stream velocity of the propellant gas

- The general exit velocity of the spacecraft

- The momentum that is imparted to the spacecraft

- The specific impulse of the spacecraft

- The total impulse of the spacecraft

12.2 Nuclear Design Parameters for Nuclear Propulsion

It is now possible to start to lay out the nuclear parameters of the ideal nuclear propulsion system that will be designed for the spacecraft. It is best to use a gaseous core open system nuclear reactor in the sample spacecraft.

Although a multitude of different reactor designs exists for spacecraft propulsion, the most feasible one is definitely the gaseous core reactor. The other types of reactors such as the solid core reactor, or the liquid core reactor are not very usable in microgravity considerations due to the number of excessive moving parts. Especially, solid core reactors are limited in their ability to withstand high amounts of heat and they usually require too much mass to be useful in long-range spacecraft. On the other hand, the selections of liquid core reactors pose stability and control problems for the spacecraft, especially in the microgravity environment of space and vacuum.

Here is a representation of gas core nuclear propulsion system that was chosen for this thesis in Figure 12.2.

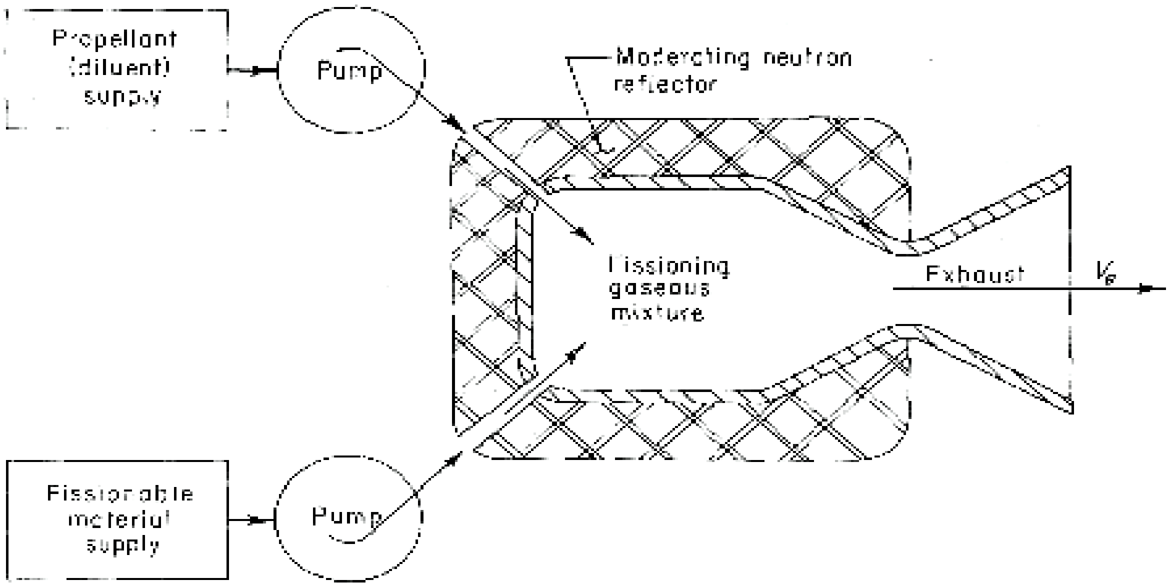


Figure 12.2: Sample Design of a Gaseous Reactor in a Nuclear Spacecraft

These main nuclear parameters can be summarized as:

- The nuclear core of the reactor in the spacecraft can be deemed to have a cylindrical geometry due to the cylindrical shape of the rocket.
- The spacecraft will run on H_2 diatomic Hydrogen slush fuel and the fuel will be stored in liquid state and transformed to gaseous state.
- The nuclear reactor will use U235 as nuclear fuel. It will be in gaseous form of Uranium Hexafluoride for stability purposes.
- For calculating a cost function, the cost of Uranium is estimated at $10000 per kilo of Uranium Hexafluoride, as per the data obtained by the International Energy Institute.
- The nuclear reactor will be in the form of an open cycle gas cored nuclear reactor.
- The reactor will have two inlets. One of the inlets will allow hydrogen to be pumped in to the reactor core (in gaseous form), while the other inlet will allow the Uranium Hexafluoride to be pumped in to the core.
- The nuclear reactor should have a diameter of 3 to 5 meters. (Turchi, 1998)

- The nuclear reactor will be shielded by a neutron moderator shield with 0.50 cm in thickness (Czysz, 2006).
- The nuclear reactor will allow for continuous pulse operation of the spacecraft
- The system reaction of the nuclear reactor in the spacecraft will be controlled by limiting the nuclear fuel and the propellant inlets (Rom and Ragsdale, 1962).
- The reactor chamber will have a single outlet and the hydrogen fuel that is heated in the reactor will be allowed to leave the nuclear reactor through this single outlet.
- The outlet will be connected to a thermodynamic convergent – divergent nozzle in the spacecraft (Taylor, 2009).
- Due to the fact that the spacecraft is designed to be working in a microgravity environment, the flow in the reactor and in the exit chamber will be thought as laminar flow. This is supported by the fact that there is no gravity or buoyancy on the flow and the thermal distribution of heat can be thought as uniform due to the geometry of the reactor. Hence, the particles are assumed to move axially in the direction of the flow (Sanchez, 2005).
- The heat transfer characteristics of the nuclear reactor will be designed as a simple conduction through a cylindrical geometry, plus molecular conduction through a flow represented by the Navier Stokes Equations in a cylindrical geometry, basic thermal radiation through a cylinder and simple ionization and disassociation of thermionic energy from the Hydrogen fuel (Bussard, 1965).
- Forced convection is assumed in the propellant flow area of the spacecraft.
- For the purposes of simplification, it’s possible to assume a finite length and infinitely long cylinder in calculations (Taylor, 2009).
- In the design process, it is assumed that only the Newton’s Third Law of Motion (The Conservation of Momentum) holds true with no friction.
- The whole system is assumed to be in a microgravity environment with no effect by the Earth’s gravity well.

- There is a semi vortex flow in the nuclear reactor core. In the flow, the Uranium is flowing at a rate that is 1000 times slower than hydrogen.
- There is 100:1 correlation in the flow. Thus, for every 100 units of hydrogen, there is one unit of Uranium in the mass flow and in the nuclear reactor core.
- The reactor is assumed to work with the minimal fuel density that is required for criticality.
- The criticality of the nuclear reactor is controlled through pressure and by the amount of fissile material present in the system. No control rods are used.
- The separation ratio of Uranium from the hydrogen during the exit phase is assumed to be less than 10^{-3} (Bussard, 1965).
- It is assumed that the hydrogen temperature is directly linked to reactor operating temperature and the stream exit velocity is proportional to temperature.

12.3 Primary Design Considerations

It is essential for the designer to be aware of the aerospace design requirements, as well as the nuclear design requirements. Perhaps the main motivational factor needs to be simplicity, since complicated designs will produce more problems in real life situations. Even the simplest reactor concept will lead to a fairly complex real reactor system and the design that starts with undue complexity may never see the light of day as a reliable power source. In addition, with the extra complicating factors of spacecraft, the system may easily become chaotic and unstable due to complex designs. It is possible to first design the spacecraft and its nuclear reactor by using operational parameters and boundary conditions. Hence, the corresponding equations can be formed to represent the system with its various modules (Delta V, Specific Impulse, Heat Transfer, Reactor Power Efficiency etc). These equations can then be modeled with various software for simulation. COSMOS-WORKS can be used for general design, FLUENT can be used for flow analysis and SPACECAD can be used for designing the rocket or the spacecraft with the appropriate parameters.

12.4 Sample Design Parameters

In the sample spacecraft that will utilize nuclear propulsion, the following parameters can be used to analyze the performance of a hypothetical nuclear spacecraft. For a sample design, some of the operational parameters (such as length, diameter, thermal dissipation factors etc) of Orion Spacecraft of NASA have been used. However, the projected parameters are originally declared within the guidelines provided in this thesis. The calculation parameters are calculated according to the equations outlined in the thesis.

Projected Parameters

Projected Specific Impulse for the Spacecraft	:	3000 sec
Projected Thrust for the Spacecraft	:	11 MN
Projected % of Payload	:	% 2
Projected % of Structure of Spacecraft	:	% 18
Projected % of Propellant & Nuclear Fuel	:	% 80

Spacecraft Parameters

Length of Spacecraft	:	6.61 m
Diameter of Spacecraft	:	5.03 m
Launch Mass	:	8485 kg
Habitable Volume	:	10.22 m^3
Crew Size	:	6 astronauts
Thermal Dissipation	:	6.3 kW

Operational Parameters

T_c = 8000 K

P_c = 45 MPa = 444 atm

P_e= 0 atm (Vacuum Pressure)

M = 4

ε = 77.5

$\Upsilon = 1.4$

$R = 8.314\,472\ J\ K^{-1}\ mol^{-1}$

Propellant Fuel = H_2 (Liquid Di-hydrogen)

Propellant Temperature $T_{propellant}$ = 14.01 K = - 259.14 C

Nuclear Parameters

Nuclear Fuel = Uranium Hexafluoride (UF_6)

Nuclear Fuel Temperature T_{fuel} = 298 K = 25 C

Mass Flow Rate of Nuclear Fuel = Mass flow of H_2 / 1000

UF_6 Pressure = 68 atm = 70.3 kg force / cm^2

Calculated Parameters

Exhaust Velocity from Equation 4.1	:	29430 m/sec
Calculated Mass Ratio (m/m_o) from Table 4.1	:	5
Delta V from Equation 4.6	:	47365.76 m/sec
Subsonic H_2 Stream Velocity from Equation 4.12	:	341.18 m/sec
Subsonic UF_6 Stream Velocity	:	0.341 m/sec
Propellant H_2 Mass Flow from Equation 4.3	:	373.77 kg/s
Nuclear Fuel UF_6 Mass Flow from Parameters	:	0.37377 kg/s
Separation Ratio from Equation 7.2	:	10^{-3}

The parameters defined above are useful for understanding the performance criteria of a nuclear propelled spacecraft such as a rocket. The mass flow as well as the velocity can be determined and the overall performance criteria will be the temperature of the reactor as well as the corresponding temperature of the propellant. Then, stream exhaust velocity and the nozzle performance can be analyzed to see how the spacecraft will behave.

By using the NASA Spacecraft Nozzle Flow Software, it is possible to see various performance criteria for the parameters specified above in Figure 12.3. The first diagram shows the output for a specific impulse of 3000 seconds and with H_2 flow of 373.77 kg/s as specified above. The nozzle flow comes out as over expanded and the temperature is assumed to be isentropic with a nuclear heat source of 8000 K from the reactor.

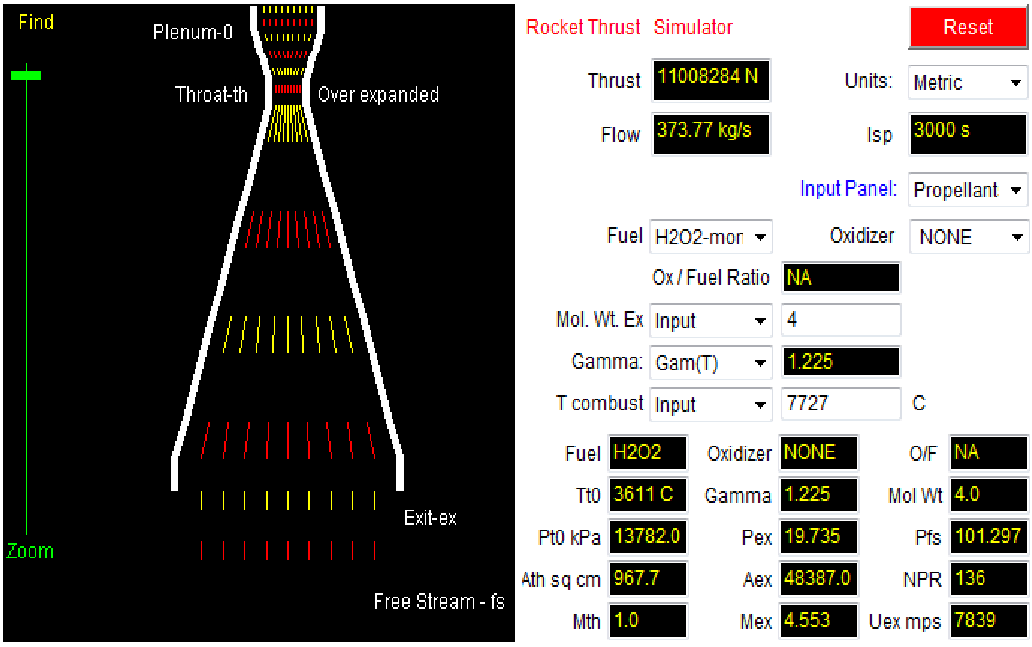


Figure 12.3 : NASA Nozzle Flow Software Output for Above Parameters

In the Figure 12.4, it is possible to see supersonic flow in the nozzle with a wide combustion area (nuclear reactor heating area) and a wider and lengthier throat for a more faster stream velocity in the flow, even with lesser temperatures. Thus, even with a different geometry of the nozzle, efficient flow characteristics are attained with isentropic nuclear heating of the propellant.

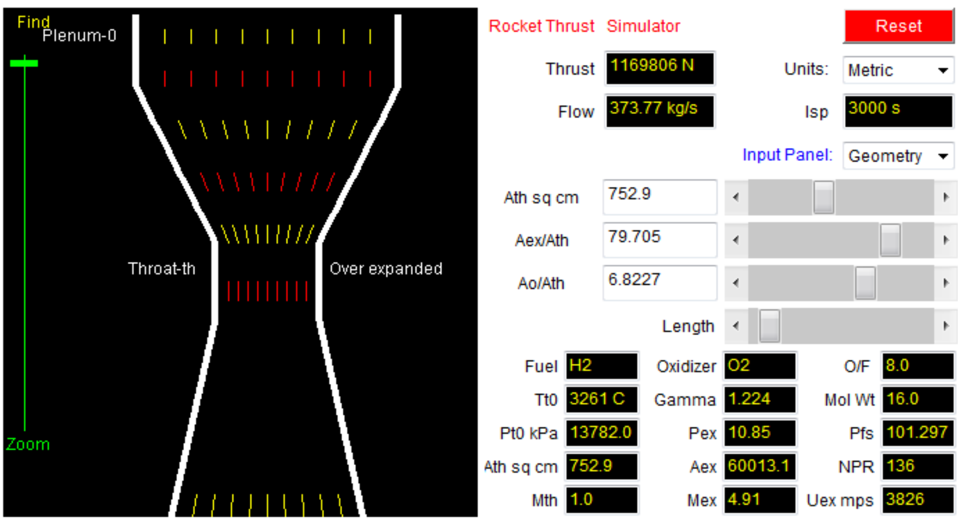


Figure 12.4 : Supersonic Nozzle Flow Output with NASA Flow Software Output

13. SOME COMPLICATIONS OF NUCLEAR PROPULSION SYSTEMS

13.1 Limitations on the Budgets of Space Programs

Americans have made considerable progress by utilizing NERVA rockets and the ROVER Program (by the work of Los Alamos National Laboratory) as well as the work of AEC / NASA Space Nuclear Propulsion Office. However, still lots of serious problems remain with the usage of nuclear propulsion. The main problem stems from the fact that there are just too many unknown variables, which depend upon each other in a stability matrix.

So far, the theoretical groundwork described in the above chapters show that there are significant developments. In fact, the developments show that these reactors are really making progress, so that interstellar travel may be possible some day. However, although the theory for using reactors for space propulsion is sound; the technological background does not provide enough data to the design engineer for a better and more stable design.

Especially after the end of the Cold War, due to the end of the competition between the West and the Soviet bloc, the developments in these fields have been funded less and less. In fact, it can be said that the most work on using nuclear reactors for space propulsion have been done in the 1960's, as both the Americans and the Soviets were competing in the arena in order to create a feasible nuclear rocket that could be used both for extra atmosphere flight, as well as for solar flight within the solar system. Especially in the 1960's, it was a matter of national pride to be the first in the race for space (Duggins, 2007).

Nowadays, with the decrease in the resources of NASA and ESA, it seems difficult that more budgets will be proportioned for the use of nuclear resources for space propulsion. In fact, even testing nuclear rockets in Earth like environment can be very costly in terms of budget and effort. Especially, with the end of the Soviet Nuclear Space program, the progress has slowed down considerably, as the race for space has become almost non-existent. Nevertheless, the fact that nuclear propulsion systems remain the only feasible means for interstellar travel makes future research compulsory.

13.2 Technological Problems with Gas Core Reactors for Space Propulsion

As mentioned above, the theoretical concepts for using gaseous core nuclear reactors for spacecraft propulsion in microgravity environment are very sound; but the necessary technology for a more efficient design should be advanced in this field. As can be seen in the preceding sections of this thesis, there is a lot of solid theory backed by Mathematics. However, the choice of materials for the necessary temperatures needs some progress made in technology.

Hence, the overall situation for advanced high thrust nuclear propulsion leaves much to be desired. There is no clear course being open without performance limitations, technical problems, or political questions. Here are some important technological considerations, which need to be handled before more serious work can be conducted on these types of projects:

- The crucial areas of material properties and heat transfer could be profitably studied experimentally at a modest expenditure to clarify further the feasibility and the limitations of these concepts.
- In gas core concepts for nuclear reactors in spacecraft, there are still many problems that are interwoven together. These problems include efficiently controlled heat transfer, fluid flow, as well as nuclear reactor criticality. These concepts need to be tested experimentally both in Earth like conditions as well as in microgravity conditions, to see how these concepts behave under certain critical boundary criteria (Czysz, 2006).
- It is characteristic for gas core rocket engines to operate at high pressures and high temperatures for the best results. This is the only way in which high specific impulses can be reached with these kinds of rockets. In fact, 1 million kgs of thrust is the minimum desired thrust power that can be beneficial for the necessary high thrust / weight ratio (Taylor, 2009).
- At low temperatures, such as below 7000 K, hydrogen is transparent to the radiation and thus sufficient high gradients cannot be reached due to this

transparency. Thus, the hydrogen in this system must be seeded with proper materials so that its opacity can be increased and that the necessary temperature gradients can be reached. The use of small particles such as carbon and other refractory materials have been studied both theoretically and experimentally for these purposes. For example, refractory materials have high boiling points and they are useful at higher temperatures. (Turchi, 1998)

- The uniform dispersion of particles in the propellant, particularly reduction of agglomerates, must be achieved on a larger scale.
- Heat generation in the reflector walls that happens due to increased neutron and gamma heating, can be of the same magnitude as fission heating in the core of conventional heat exchanger reactors. Unfortunately, the heating peaks at inner surfaces and this is obviously an unfavorable condition for achieving high temperatures of the gas with inward radial propellant flow.
- The propellant has to enter the cavity at high temperatures and this can cause an important engineering difficulty (Turchi, 1998).
- When a nuclear rocket is started, the initialization of the nuclear rocket can be between 15 to 20 seconds. During these seconds, two-phase flow of the hydrogen will exist in the nuclear reactor chamber. This can cause pressure instabilities and local hotspots in the reactor region (Ellerbrock and Livingood, 1962).
- Shutdown requirements can pose problems for the nuclear spacecraft, since the reactor will continue to be a heat source even after the mass flow is reduced to initiate a shutdown (Ellerbrock and Livingood, 1962).
- During both start up and cool down phase, there will be a two-phase flow from turbulent to laminar and laminar to turbulent flow. This can cause certain instabilities in the system and it can introduce new neutronics control problems for the spacecraft and the reactor (Ellerbrock and Livingood, 1962).
- Critical mass requirements can also pose serious challenges to the designer. The difficulty for this stems from the fact that the critical mass for reflector moderated reactors is quite sensitive to the detailed composition and configuration.

- As the fuel region becomes thicker to neutrons, the value in nuclear reactivity of additional fuels goes down (Turchi, 1998)
- Increased fuel density can cause higher opacity and thus this can also lead to higher temperatures in the fuel as well as higher pressures in the system. All of these present serious engineering problems to the shells, pumps, nozzles and other components in the spacecraft (Turchi, 1998).
- The corrosive properties of the propellant in the nuclear reactor system as well as the nozzle and the chamber flow system are formidable. Special composite materials need to be used, so that the necessary conditions are met both for the nuclear parts of the spacecraft as well as for the normal space flight parts. Sometimes what works for one purpose might not work for the other purpose and this can present an important engineering feat. Moreover, the designer may face other important boundary conditions of the materials (Marjon, 2002).
- Radiation damage to the materials of the spacecraft is also a formidable engineering problem that needs to be addressed. It is important for the spacecraft to be durable for long-term use and radiation can shorten the lifespan of many components (Czysz, 2006).
- It is also essential for the designer to make sure that the overall cost of the spacecraft is feasible. This means that in some cases, technologically feasible solutions will need to be overwritten in favor of more cost feasible solutions that may be available to the designer. Hence, the optimum design parameters may not always reflect the best design, but rather the cheapest possible design.

13.3 The Loss of Nuclear Fuel in Gas Core Reactors for Space Propulsion

Unfortunately, in gaseous core rockets or in spacecrafts with gaseous cores, the separation of the nuclear fuel with the propellant of the spacecraft is important. This stems from the fact that these gaseous cores contain uranium fuel, as well as hydrogen propellant in gaseous form due to the temperature and the pressure requirements of the system. This can cause a serious problem, as some nuclear fuel will be bonded and discharged through the nozzle with the propellant.

This can be very disturbing, since hydrogen would cost somewhat around 50 cents to $1 per kilogram, while processed gaseous uranium would cost around $10,000 per kilogram. For example, assume that there is %10 loss of uranium fuel with the normal flight operations of the spacecraft. This can come to mean that if 10,000 kilograms of propellant is used in the spacecraft, this may lead to lose as much as 1000 kilograms of nuclear fuel (Turchi, 1998)

Losing approximately 1000 kilograms of nuclear fuel can cost a significant amount, since gaseous uranium is approximately $10,000 per kg. This comes to a total of $10 million dollars per mission and this is a serious limitation for the mission project planner, as no one would want to lose that much money on a single mission. Table 13.1 shows the fuel economics and the costs per the separation ratio.

Table 13.1 : Cost of Nuclear Fuel as Dependent on the Separation Ratio

Separation Ratio	Fuel Mass per mission	$ Fuel Cost/ 10^6
100	10^4	**50**
1000	10^3	**5**
10,000	10^2	**0.5**

13.4 Shielding Problems for Fission Reactors in Spacecraft

One of the main problems that deals with using nuclear propulsion is how to use it in a crewed spacecraft that has astronauts on board. As can be realized from the preceding sections, there are many powerful sources of hazardous radiation that is emitted during the fission process in the spacecraft.

These shielding problems involve complex neutron and nuclear physics and thus it can be more difficult to shield against all eventualities. Moreover, in a spacecraft the problems are even more compounded. The design engineer also has to limit the amount of mass in the spacecraft, so that the necessary payload can be included. Hence, as compared to terrestrial applications, the shielding problems for the spacecraft using nuclear propulsion are more numerous and thus the designer faces serious technological design challenges. However, with new composite materials, perhaps better shielding with less mass can be utilized in the future.

Thus, some serious work needs to be performed on the shielding problems of the spacecraft. The shield should be strong enough to protect the human crew and the payload, while it should have less mass and its thickness should be as little as possible to conserve the space in the spacecraft.

Perhaps, this is the most serious design problem of the engineer, as he will have to balance safety with the cost requirements of the mission. Especially in instances where long distance missions are concerned, the design engineer will have to make sure that the space for the crew, the payload, as well as for the logistical supplies will need to be maximized. This means less space for the reactor as well as the shielding. This can especially become problematic for long-range missions, as the radiation exposure of the human crew to both the cosmic radiation and to nuclear radiation can pose long term health problems.

For example, if there is not enough volume for the air of the crew and if there is not enough space for the necessary food and the equipment, then the usage of nuclear propulsion systems will be meaningless. Thus, the requirements for cost, for the duration of the mission and the operating parameter of the mission will need to be clearly defined and juggled by the design engineer. With the advancements in technology, it is expected that this will become easier in the future. A possible futuristic design of a nuclear spacecraft is seen in the diagram below at Fig. 13.1.

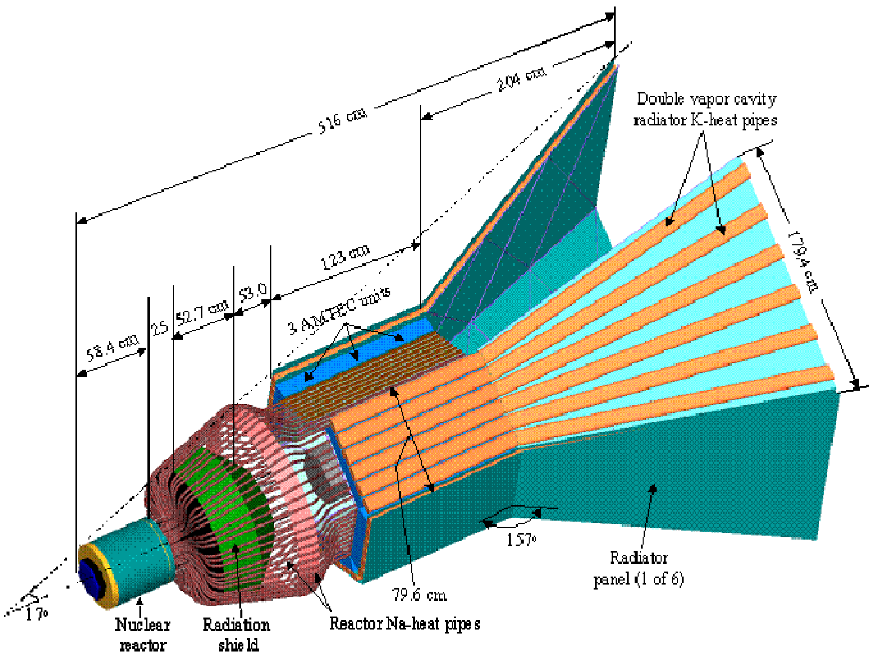


Figure 13.1 : Futuristic Design of a Nuclear Craft from University of New Mexico

14. NUCLEAR PROPULSION IN LONG DISTANCE EXPLORATION

14.1 The Prospects for Travel to the Moon with Nuclear Propulsion

There are many different prospects for space exploration by using these advanced nuclear engine concepts. As mentioned in the beginning of this thesis, one of the most important criteria for space exploration is the specific impulse. For any spacecraft to be able to go far in the solar system, it is essential for that spacecraft to have a high specific impulse.

It is obvious that these specific impulses cannot be reached with standard chemical or solid propulsion techniques as mentioned in many parts of this thesis. The specific impulse by these means of propulsion is quite small and thus as a result, missions with far reaching distances can not be contemplated. Moreover, it can be difficult to store large amounts of solid or liquid propellant for long periods of time in a spacecraft. According to a 2005 survey report by NASA, using nuclear engines can open the way for travels as far as Neptune in the future as can be seen in Fig. 14.1.

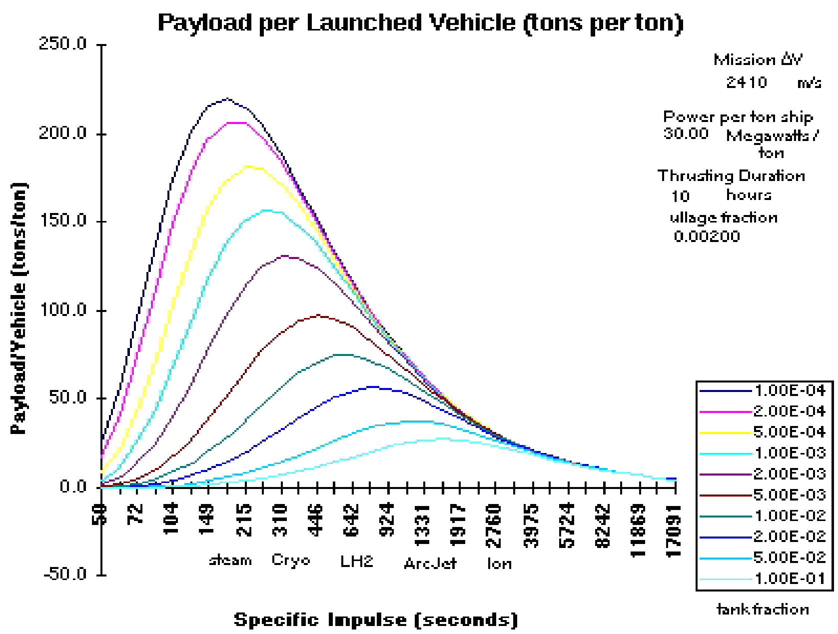


Fig 14.1: Comparisons of Payload and Specific Impulse for Moon Missions

The space exploration initiative (SEI) which has been outlined by President Bush calls for a permanent return to the moon in the early years of the 21^{st} century. According to the same Space Exploration Initiative, this is to be followed by a journey to Mars in years as early as the year 2025. Therefore, there are near plans for exploring the solar system, as the humankind intends to travel to the Moon and the planet Mars (Duggins, 2007).

Also, further plans calls for establishing and sustaining a permanent lunar outpost. This requires efficient and re-usable space transportation system for moving both the humans and the cargo between Earth – Lunar space. The Nuclear Thermal Rocket represents the best possible solution in propulsion technology, as it is ideally suited to perform piloted missions that carry cargo and humans. Operating in a combined mode, with more than twice the specific impulse of chemical propulsion spacecraft, the NTR can easily deliver and return sizable payloads from Earth to the Moon (Singh, 2003).

Hence, using nuclear reactor operated thermal rockets between Earth and the Moon would provide the necessary background for any future Mars mission. It would also provide the necessary operating experience that is helpful in serving as a technology proving ground for any Mars mission. In fact, currently, a lunar mission employing a re-usable NTR is under serious study by NASA. Major system components would be launched to Low Earth Orbit (LEO) by a heavy lift launch vehicle. Then, after a system checkout in the Low Earth Orbit, the hydrogen propellant flow would begin as the reactor is started. Then this reactor would heat up the hydrogen fuel in the spacecraft (Singh, 2003).

When the hydrogen is being heated, it would start to expand from the engine nozzle. This expansion of the hydrogen in the nozzle would produce thrust due to thermodynamic properties of gas expansion. Hence, as this hydrogen would expand and exit through the nozzle, the thrust would be maintained and the spacecraft would begin to move toward the moon.

After approximately 30 minutes of operation, the reactor would be turned off and the spacecraft would travel to the moon on the principal of momentum from the Newton’s equations of motion. The reactor would be turned on again for about 10 minutes to propulsive brake into lunar orbit. The payload would then leave the NTR vehicle and then it would be descended in to the lunar surface to perform the relevant mission. Hence, after docking, the entire lunar vehicle could be used to return to earth orbit again and thus these vehicles could be re-usable for many times. Hence, the usage of a Nuclear Thermal Rocket in between the Lunar – Earth space can be real advantageous as seen in Figure 14.2.

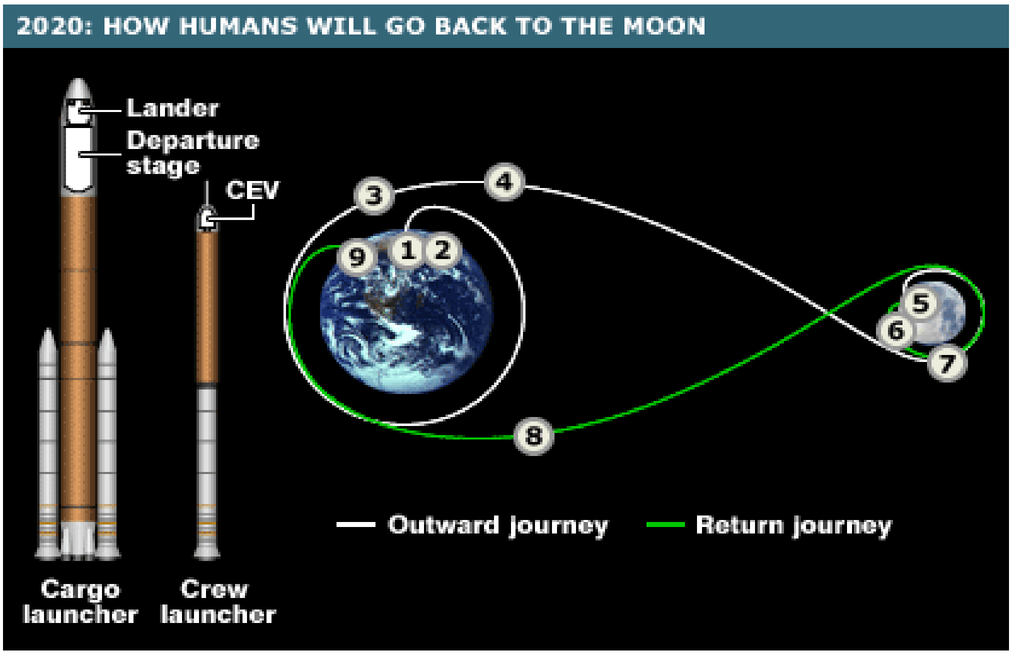


Figure 14.2: A Proposed Manned Lunar Mission with NTR

(1) A heavy-lift rocket blasts off from Earth carrying a lunar lander and a "departure stage"

(2) Several days later, astronauts launch on a separate rocket system with their Crew Exploration Vehicle (CEV)

(3) The CEV docks with the lander and departure stage in Earth orbit and then heads to the Moon with the aid of nuclear propulsion.

(4) Having done its job of boosting the CEV and lunar lander on their way, the departure stage is jettisoned

(5) At the Moon, the astronauts leave their CEV and enter the lander for the trip to the lunar surface

(6) After exploring the lunar landscape for seven days, the crew blasts off in a portion of the lander

(7) In Moon orbit, they re-join the waiting robot-minded CEV and begin the journey back to Earth again using nuclear propulsion.

(8) On the way, the service component of the CEV is jettisoned. This leaves just the crew capsule to enter the atmosphere

(9) A heat shield protects the capsule; parachutes bring it down on dry land, probably in California.

In the stages above, the craft that takes the astronauts from the outer atmosphere can be a nuclear propulsion rocket as described in the phases 3 and 7. It is also worth mentioning that it can be less costly this way.

14.2 The Prospects for Travel to Mars with Nuclear Propulsion

It is also possible to use these same concepts to travel to Mars using nuclear reactor powered space vehicle propulsion. The most efficient and doable proposals to get a crew to and from Mars safely, efficiently and relatively quickly are nuclear-powered mission profiles. Plenty of NASA astronauts would volunteer for a 2-year round trip Mars mission, but they are more hesitant to consider trip durations that stretch much beyond that, such as the durations encountered with chemically propelled missions.

A fast mission using nuclear thermal rockets could get astronauts to Mars in as little as 4 months. It would allow them a one- to two-month stay on the planet and then bring them back to Earth on a return leg that would take about eight months, getting them home in just over a year.

One of the great added strengths of the Nuclear Thermal Rocket is that it can be used to generate not only thrust, but all the power that a crew needs during interplanetary travel. Once the crew-transfer vehicle escapes from Earth orbit and reaches speed on its trip to Mars, the engines are brought down to an idle state.

Their heat is routed through a generator to produce power for crew survival, high data-rate communications, and even a cooler unit to keep the liquid hydrogen fuel from boiling off into space. Because liquid hydrogen boils at minus 217 degrees Celsius, the loss of hydrogen propellant is a serious problem which forces most mission designers to carry a great deal of extra propellant to make up for the loss.

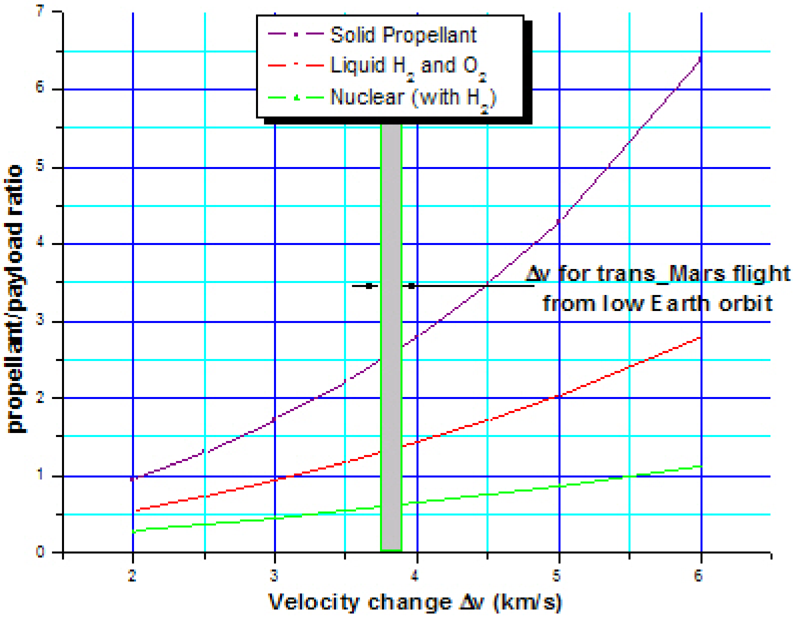


Figure 14.3: Comparisons of Velocity Change and Momentum Transfer in Mars Mission with Nuclear and other methods

With nuclear reactors, though, there is plenty of energy to run a refrigeration system to keep the hydrogen cold. This greatly reduces the total mass of the vehicle. Nuclear reactors even provide enough power to create artificial gravity, a feature that should protect the astronaut crew from the physiological ravages of living in low-gravity conditions for extended periods. Hence, the best specific impulse and exhaust velocities and conditions are reached with nuclear reactor powered space vehicles. The advantage of a Mars Mission can be seen clearly in Figures 14.3 and 14.4.

Figure 14.4 : Momentum Transfers in Mars Mission Using Nuclear Propulsion

In time, NASA hopes to spend more money on future Mars projects using nuclear propulsion. Some of these are based on theories, but the technological developments concerning nuclear spacecraft propulsion is catching up. It is recommended that any nation, that hopes to go to space some day, start the necessary technological work relating to nuclear propulsion in order to be ready for the future.

14.3 The Prospects for Solar System by Nuclear Propulsion

The prospect of using nuclear propulsion for spacecraft is filled with numerous possibilities. For travels that involve only the solar system, the possibility of nuclear propulsion is important, because even at relativistic speeds, the crew will have to spend considerable time travelling. Thus, with the high specific impulse of the nuclear engines, there is the possibility of the spacecraft to have a chance to reach the far reaches of the solar system in a reasonable amount of time. Here is a diagram at Table 14.1 that shows the speed and the distances, as well as the travel times to various places in the solar system.

Table 14.1: Travel Times within the Solar System (NASA)

Object	Mass	Diameter	Distance	Time at c from the Sun	Time at V_{escape}
Sun	332,946	109.0	0.00		
Mercury	0.060	0.38	0.30	2.493 minutes	132.018 days
Venus	0.082	0.95	0.72	5.984 minutes	142.018 days
EARTH	**1.000**	**1.00**	**1.00**	8.311 minutes	0.000 days
Mars	0.110	0.53	1.52	12.633 minutes	215.87 days
Asteroids			2.70	22.440 minutes	1.050 years
Jupiter	317.80	11.20	5.20	43.218 minutes	2.022 years
Saturn	95.17	9.40	9.54	1.321 hours	3.709 years
Uranus	14.60	4.20	19.18	2.657 hours	7.458 years
Neptune	17.25	4.00	30.05	4.162 hours	11.684 years
Pluto	0.100	0.50	39.40	5.458 hours	15.320 years
Kuiper Belt	40.0		30 to 50	5.541 hours	15.553 years
Heliopause			100.00	254.0 days	38.883 years

Note: 1 AU = 1.496×10^8 km (AU) average.

In the first stage of the project, it is aimed to use nuclear thermal propulsion systems on probes and satellites within the solar system. With automated Nuclear Thermal Rockets, emergency unmanned missions can be easily carried out anywhere within the solar system. In time, it will also be possible to travel with manned craft to anywhere within the solar system by using nuclear propulsion.

14.4 Interstellar Travel by Nuclear Propulsion

Especially when thinking of travelling to interstellar distances, then utilizing nuclear means of propulsion is the only way to proceed. Once the necessary momentum with the nuclear engines is acquired, the spacecraft can easily continue to travel until it reaches its destination. In addition, the nuclear reactor in the spacecraft can be used to supply electricity for prolonged periods of time until the mission is completed. Thus, both for the power needs as well as for the propulsion requirements; using nuclear power is the only technologically available solution for interstellar exploration projects in the future.

14.5 The Prospects of Using Fusion for Nuclear Propulsion

The best form of nuclear propulsion for interstellar spacecraft has been idealized as a fusion engine. In a fusion reactor, tritium and deuterium (two isotopes of hydrogen) are combined through a nuclear fusion process, so that helium is formed. During this fusion process, some mass is decreased, energy is released, which is expressed as heat and radiation as seen in Figure 14.5. (Williams, 1997).

The good thing about fusion reactors is the fact that the interstellar space is filled with lots of heavy hydrogen atoms (even in the void of space) and thus there is not a need to stock large amounts of fuel for the interstellar journey. This would leave more room for the astronauts, as well as for the payload that the spacecraft is carrying. Hence, there is no need to store any propellant (as some missions can require hundreds of thousands of tons of propellant for long distances and this can be costly in terms of money and space.) Perhaps with fusion technology, missions that take many years to complete could become possible (Williams, 1997).

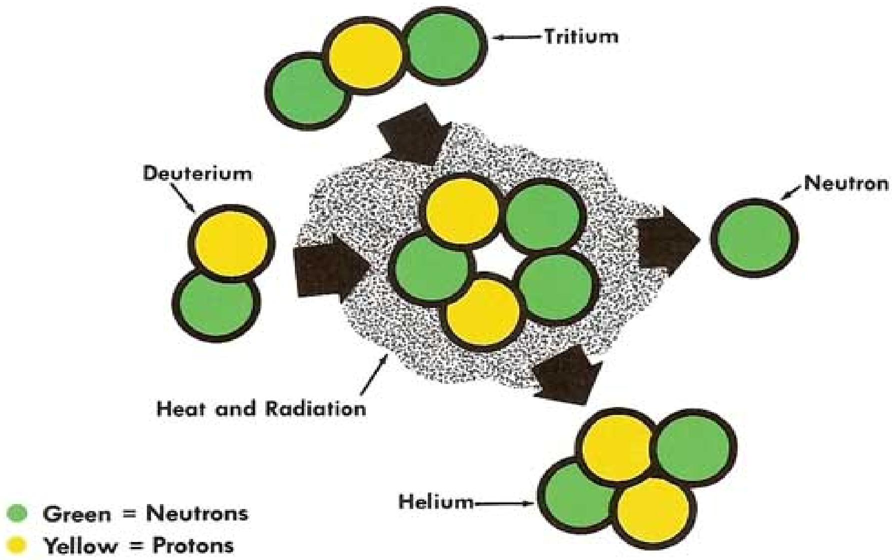


Figure 14.5: Typical Fusion Reaction

14.6 Means of Propulsion by Advanced Nuclear Technology

The prospect for interstellar travel lies in both fission reactors as well as in fusion reactors that can be used to power up spacecraft in interstellar space. In time, perhaps more exotic forms of propulsion such as by using anti-matter and matter can be possible, but that time still lies far away in the future. The Figure 14.6 below shows clearly why the prospects of using fission reactors and fusion reactors are important for the future of space travel.

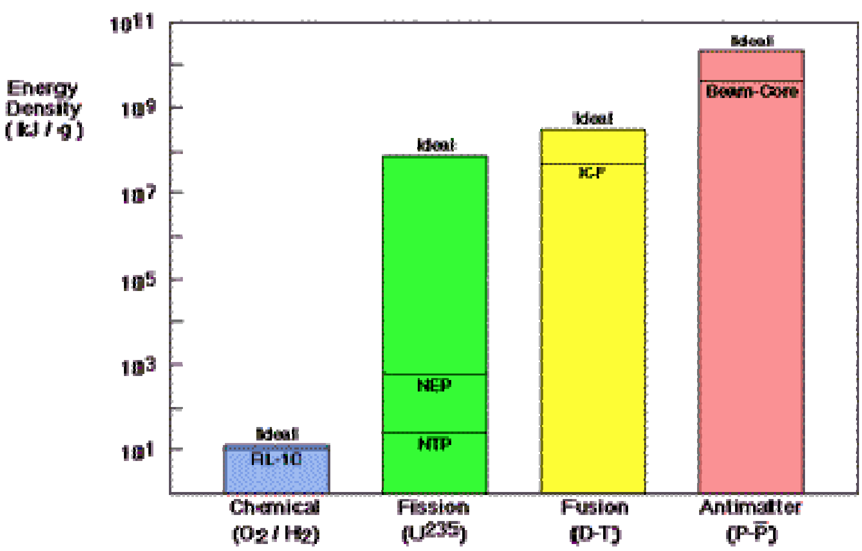


Figure 14.6 : Comparisons of Advanced Systems for Space Propulsion

15. CONCLUSION

In any case, this thesis recommends that prospects for space propulsion by using gas core reactors is analyzed further and that more exotic ways such as fusion reactor propulsion are also investigated. Even more exotic means of nuclear propulsion systems are proposed by using anti-matter drives capable of propelling the spacecraft by the annihilation energy of matter and anti-matter.

Especially, the fact that gas core nuclear propulsion is feasible with the current technology is an important fact. With the advancements in the magneto-dynamic confinement of the propellant – uranium mixture, it is possible to ionize the uranium and control it with magnetic fields. Thus, as discussed above, it will be possible to use gas core nuclear rockets with high specific impulse for Moon and Mars missions. Unfortunately, the budget of both leading countries that is spent on Nuclear Space Programs has dwindled a lot. However, it is the belief of this thesis that more researchers in the academia have to work on these nuclear propulsions concepts to make up for that deficit. ***Even for countries without space programs like Turkey, working on nuclear propulsion techniques can help secure the future***.

In time, the technology will catch up with the theory and it is essential to be prepared when that time comes. Whether it has a space program or not, any nation which is a signatory to United Nations Peaceful Expansion and Development of Space Program by UNOOSA should take part in academic research concerning nuclear propulsion. Thus, by further research into nuclear propulsion, the prospects for solar system travel as well as for interstellar travel will become possible in the future, and the mankind can take his rightful place among the stars.

REFERENCES

Aftergood, Steven, 1989. Background on Space Nuclear Power, Science and Global Security pp 93-107, Volume 1, New York

Bogart, Donald and Lantz, Edward, 1963. Nuclear Physics of Solid Core Gas Cooled Propulsion Reactors, NASA Lewis Research Center Classified Report, United States Government Printing Office, Washington D.C

Bolonkin, Alexander, 2008. Simplest AB Thermonuclear Space Propulsion and Electric Generator, International Journal of Aircraft Engineering and Aerospace Technology, Volume 80/1, pp 11-17, New York

Borowski, S. K., Dudzinke, L., Mcguire, M., Gilland James H. 2001. Bimodal Nuclear Thermal Rocket Propulsion for a Hope Mission to Callisto, 52 International Astronautical Congress, IAA-01-IAA-01-IAA.13.3.05, Toulouse, France.

Bussard, R..W and Delauer, R.D., 1965. Fundamentals of Nuclear Flight, Mc Graw Hill, New York.

Bussard, R.W and Delauer, R.D., 1958. Nuclear Rocket Propulsion, Mc Graw-Hill, New York.

Butler, C. and Albright, D., 2007. Core Physics and Kinetics Calculations for the Fissioning Plasma Core Reactor, NASA White Paper, NASA/CR-2007-214724

Campbell, W.E. et al., 1967. NPSP Selection for A Nuclear Rocket, Paper 67-476, Collection of Technical Papers on Nuclear Propulsion, AIAA 3rd Propulsion Joint Specialist Conference, Washington D.C

Clark, J.S., 1992. An Historical Collection of Papers on Nuclear Thermal Propulsion, AIAA, Washington, D.C.

Czyss, Paul A. and Bruno Claudio, 2006. Future Spacecraft Propulsion Systems : Enabling Technologies for Space Exploration, Praxis Publishing, Berlin

Duggins, Pat, 2007. Final Countdown – NASA and the End of the Space Shuttle Program, University Press of Florida, Gainesville.

Ellerbrock, Herman H., Livingood, John N., Straight, David M., 1963. Fluid Flow and Heat Transfer Problems in Nuclear Rockets, NASA Lewis Research Center Classified Report, United States Government Printing Office, Washington D.C.

El-Genk, M.S., 1994. A Critical Review of Space Nuclear Power and Propulsion 1984-1993, AIP Press, New York

Finseth, J.L. 1991. ROVER Nuclear Rocket Program: Overview of Rover Engine Tests Final Report, NASA George C. Marshall Space Flight Center, File No: 313-002-91-059

Grayamov, G.M., "Nuclear Power for Space Vehicles: A New Dimension in the Energy Technology of the Future" Publication of Iavestiya Academy USSR Vol 29 pp-24-33 1991 USSR Academy of Sciences Presentation 1990.

Heister, Steve, 1998. Nuclear Thermal Propulsion, University of Purdue, School of Aeronautics and Astronautics Course Notes

Hill, P. and Peterson, C., 1992. Mechanics and Thermodynamics of Propulsion, 2nd Edition, Addison Wesley, Reading, Massachusetts.

Himmel, S.C., Dugan, J., Luidens, R.W., Weber, R. J., 1961. A Study of Manned Nuclear Rocket Missions to Mars, Paper 61-49, Institute of Aerospace Sciences

Himmel, S.C., Dugan, J., Luidens, R.W., Weber, R. J., 1961. Nuclear Rocket Vehicles for Mars and Venus Missions, ARS/ORNL Space Nuclear Conference, Tennessee, May.

Johnson, Paul G. and Rom, Frank E., 1962. Perigee Propulsion for the Orbital Launch of Nuclear Rockets, NASA TR R-140

Kulcinski, G.L., 1997. Nuclear Power in Space, NEEP602 Course Lecture Notes, University of Wisconsin

Knight, B.W, McInteer, B.B., Potter, R. M., Robinson, E.S., 1957. A Metal Dumbo Rocket Reactor, Los Alamos Scientific Laboratory Report to the U.S. Atomic Energy Commission No: W-7405-ENG 36

Litchford, Ron J., Foote, John P., Hickman, Robert, 2005. Long Duration Hot Hydrogen Exposure of Nuclear Thermal Rocket Problems, NASA Marshall Space Flight Center Unclassified Report, Washington D.C.

Marjon, P, 2002. Space Maintenance of Nuclear Rocket Propulsion Vehicle, IEEE Transaction of Aerospace Conference Brief, Florida

Mohler, Ronald, 1964. Stability and Control of Nuclear Propulsion, Los Alamos Scientific Laboratory Unclassified Report, California.

Norring, Robert, 2004. Optimum Utilization of Fission Power with Gas Core Reactors, M.Sc. Thesis, University of Florida, Gainesville, Florida.

Mohler, R. and Perry, J., 1961. Nuclear Rocket Engine Control, *Nucleonics* Vol. 19, No:4 April, p 80-84

Pike, J.,1998. Nuclear Thermal Propulsion Concepts, http://www.fas.org/nuke/space/index.html, World Wide Web,

Rom, Frank E., 1959. Advanced Reactor Concepts for Nuclear Propulsion, Astronautics, Volume 4, No 10, pp 20-22

Rom, Frank E., Lietzke, Armin F., Johnson, Paul G., 1962. Small Nuclear Rockets for Space Missions, *Nucleonics*, Volume 20, No:11

Rom, Frank E., Ragsdale, Robert G., 1962. Advanced Concepts for Nuclear Rocket Propulsion, United States Government Printing Office, Washington D.C.

Rosen, Robert and Schnyer Dan, 1989. Civilian Uses of Nuclear Reactors in Space, Science and Global Security, Washington D.C.

Sanchez, Manuel, 2005. Non Equilibrium Flows in Rocket Propulsion, MIT Lecture Notes on Rocket Propulsion, Massachusetts.

Sanders, John C., Heppler, Herbert J., Hart. Clint E. 1963. Problems in Dynamics and Control of Nuclear Rockets, NASA Lewis Research Center Classified Report, United States Government Printing Office, Washington D.C.

Sidi, Marcel J., 2002. Spacecraft Dynamics and Control – A Practical Engineering Approach, Cambridge University Press, Cambridge.

Singh, Gurkirpal, 2003. Best Fit Conic Approximation of Spacecraft Trajectory, NASA Tech Brief, NPO-40622, California

Sutton, G., 1992. Rocket Propulsion Elements, 6th Edition, John Wiley and Sons, New York.

Taylor, Travis S., 2009. Introduction to Rocket Science and Engineering, CRC Press, New York.

Thring, M.W., 1960. Nuclear Propulsion, Butterworth Press, London.

Thom, Karlheinz and Schwenk, Francis C., 1977. Gaseous Fuel reactor Systems for Aerospace Applications, AIAA Conference on the Future of Aerospace Power Systems, March 1-3, St. Louis.

Turchi, Peter J., 1998. Propulsion Techniques Action and Reaction, AIAA American Institute of Aeronautics and Astronautics

Wiliams, C.H. and Borowski S.K., 1997. An Assessment of Space Propulsion Fusion Concepts and Desired Operating Parameters for Fast Solar System Travel, AIAA Paper July

Wiliams, C.H. and Borowski S.K., 1998. A Spherical Torus Nuclear Reactor Space Propulsion Vehicle Concept, AIAA 34th Joint Propulsion Conference

Witter, Jonathan, 1993. Modeling for the Simulation and Control of Nuclear Reactor Rocket Systems, Massachusetts Institute of Technology P.hD Thesis, Massachusetts

URL 1, http://www.nasa.gov accessed at 12/02/2009

URL 2, http://www.generaldynamics.com accessed at 01/06/2009

APPENDICES

APPENDIX A.1 : The Characteristic Properties of Hydrogen
APPENDIX A.2 : The Characteristic Properties of Uranium
APPENDIX A.3 : The Characteristic Properties of Uranium Hexafluoride

APPENDIX A.1

PROPERTIES OF HYDROGEN

Appearance	
colorless gas	
General properties	
Name, symbol, number	hydrogen, H, 1
Element category	nonmetal
Group, period, block	1, 1, s
Standard atomic weight	1.00794(7) $g \cdot mol^{-1}$
Electron configuration	$1s^1$
Electrons per shell	1 (Image)
Physical properties	
Color	colorless
Phase	gas
Density	(0 °C, 101.325 kPa) 0.08988 g/L
Melting point	14.01 K-259.14 ° ,C-434.45 ° ,F
Boiling point	20.28 K-252.87 ° ,C-423.17 ° ,F
Triple point	13.8033 K (-259°C), 7.042 kPa
Critical point	32.97 K, 1.293 MPa
Heat of fusion	(H_2) 0.117 $kJ \cdot mol^{-1}$
Heat of vaporization	(H_2) 0.904 $kJ \cdot mol^{-1}$
Specific heat capacity	(25 °C) (H_2) 28.836 $J \cdot mol^{-1} \cdot K^{-1}$

Vapor pressure

P/Pa	1	10	100	1 k	10 k	100 k
at T/K					15	20

Atomic properties	
Oxidation states	1, -1 (amphoteric oxide)
Electronegativity	2.20 (Pauling scale)
Ionization energies	1st: 1312.0 $kJ \cdot mol^{-1}$
Covalent radius	31±5 pm
Van der Waals radius	120 pm
Miscellanea	
Crystal structure	Hexagonal
Magnetic ordering	diamagnetic[1]
Thermal conductivity	(300 K) 0.1805 $W \cdot m^{-1} \cdot K^{-1}$
Speed of sound	(gas, 27 °C) 1310 m/s
CAS registry number	1333-74-0

APPENDIX A.2

PROPERTIES OF URANIUM	
Name, symbol, number	uranium, U, 92
Element category	actinide
Group, period, block	n/a, 7, f
Standard atomic weight	238.02891(3) g·mol^{-1}
Electron configuration	[Rn] 5f^3 6d^1 7s^2
Electrons per shell	2, 8, 18, 32, 21, 9, 2 (Image)
Physical properties	
Phase	solid
Density (near r.t.)	19.1 g·cm^{-3}
Liquid density at m.p.	17.3 g·cm^{-3}
Melting point	1405.3 K1132.2 ° ,C2070 ° ,F
Boiling point	4404 K4131 ° ,C7468 ° ,F
Heat of fusion	9.14 kJ·mol^{-1}
Heat of vaporization	417.1 kJ·mol^{-1}
Specific heat capacity	(25 °C) 27.665 J·mol^{-1}·K^{-1}

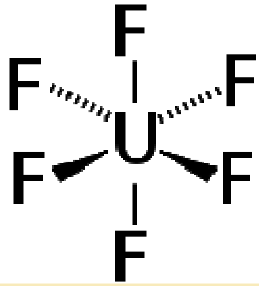
Vapor pressure

P/Pa	1	10	100	1 k	10 k	100 k
at *T*/K	2325	2564	2859	3234	3727	4402

Atomic properties	
Oxidation states	**6**, 5, 4, 3[1] (weakly basic oxide)
Electronegativity	1.38 (Pauling scale)
Ionization energies	1st: 597.6 kJ·mol^{-1} 2nd: 1420 kJ·mol^{-1}
Atomic radius	156 pm
Covalent radius	196±7 pm
Van der Waals radius	186 pm
Miscellanea	
Crystal structure	Orthorhombic
Magnetic ordering	paramagnetic
Electrical resistivity	(0 °C) 0.280 µΩ·m
Thermal conductivity	(300 K) 27.5 W·m^{-1}·K^{-1}
Thermal expansion	(25 °C) 13.9 µm·m^{-1}·K^{-1}
Speed of sound (thin rod)	(20 °C) 3155 m/s
Young's modulus	208 GPa
Shear modulus	111 GPa
Bulk modulus	100 GPa
Poisson ratio	0.23
CAS registry number	7440-61-1

APPENDIX A.3

PROPERTIES OF URANIUM HEXAFLUORIDE	



Identifiers	
CAS number	7783-81-5 ✓
UN number	2978 (<1% ^{235}U) 2977 (>1% ^{235}U)
RTECS number	YR4720000
Properties	
Molecular formula	UF_6
Molar mass	352.02 g/mol
Appearance	colorless solid
Density	5.09 g/cm^3, solid
Melting point	64.8 °C (triple point)
Boiling point	56.5 °C (sublimes)
Solubility in water	Reacts
Solubility	soluble in chloroform, CCl_4, liquid chlorine and bromine dissolves in nitrobenzene
Structure	
Crystal structure	Orthorhombic, oP28
Space group	Pnma, No. 62
Coordination geometry	octahedral (O_h)
Dipole moment	0
Thermochemistry	
Std enthalpy of formation $\Delta_f H^\circ_{298}$	−2317 kJ/mol
Standard molar entropy S°_{298}	228 J K^{-1} mol^{-1}
Hazards	
MSDS	ICSC 1250
EU Index	092-002-00-3
EU classification	Very toxic (**T+**) Dangerous for the environment (**N**)
R-phrases	R26/28, R33, R51/53
S-phrases	(S1/2), S20/21, S45, S61
Flash point	Non-flammable

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Publications

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The Future of Space Travelling (2005, Associated Content)
The Prospects for Interstellar Exploration (2005, Associated Content)
Harnessing Space Power (1992, Space Science Newsletter, Florida, May Ed.)

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