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Low Enriched Uranium based Nuclear Rocket Propulsion Technology: Mars Exploration Mission

ORIGINAL ARTICLE

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Abstract - Many space agencies like NASA, SPACE-X have promised to send humans into the red planet in future. So, considering their project of mars colonization, nuclear rocket propulsion would be the better option. Replacing chemical rockets by nuclear rockets may reduce the mission duration and also can reduce the mass of the propellant used. In chemical rockets, propellant releases energy through combustion but in case of nuclear rockets, propellant i.e. hydrogen is heated up in controlled fission reaction in nuclear reactor inside the rocket engine. Specific impulse of the nuclear rocket is greater than chemical rocket. This helps in providing gigantic thrust as a result mission duration is decreased. The challenging parameter of increasing specific impulse is solved by maximizing specific impulse which is done by increasing the exhaust core temperature. The fuel is selected in such a way so that the exhaust temperature would be obtained. The (U, Zr) C – graphite fuel is selected because it has high uranium density and melting point is equivalent to exhaust core temperature which is sufficient enough to enhance the reactivity of the fissile material and thus to increase the rocket performance. A mathematical analysis shows that the percentage of mass of propellant used in mars mission will be lesser than the chemical rockets because the specific impulse is expected to be more in nuclear propulsion. The specific impulse obtained from the CFD Analysis of rocket nozzle is 979 sec with exit velocity of 9604m/s.

Keywords: *Nuclear; Propulsion; Rocket; Nozzle; CFD*

I. INTRODUCTION

The concept of Nuclear Rocket Propulsion arises if we talk about long distanced mission like sending humans to mars or to asteroids. Nuclear energy is one of the most enabling as well as proposed technology in space exploration. Basically, Nuclear Thermal Propulsion (NTP) consists of liquid hydrogen as propellant nuclear reactor & nozzle. Unlike conventional chemical rockets, it uses the nuclear reactor that heats the propellant & then expands through the nozzle. Recently, we are focusing on the development of engines with targeted specific impulse 1000sec approx. In order to achieve this value, the nuclear reactor should heat a specified flow rate of liquid hydrogen to a temperature of 2700K. The reactor operating temperature must not exceed the melting points of surrounding rocket components.

Conceptually, the NTP systems is remarkably simple; but control of nuclear fission reaction is somehow challenging. This is minimized with the help of control drums positioning. The Research and development of Nuclear Thermal Propulsion has been already done by USA & Russia under ROVER and NERVA (Nuclear Engine for Rocket Vehicle Applications) programs. [2] In this design model of NTP, the liquid hydrogen is allowed to plunge and heated up in Nuclear reactor with the help of turbine pump as shown in fig. 1

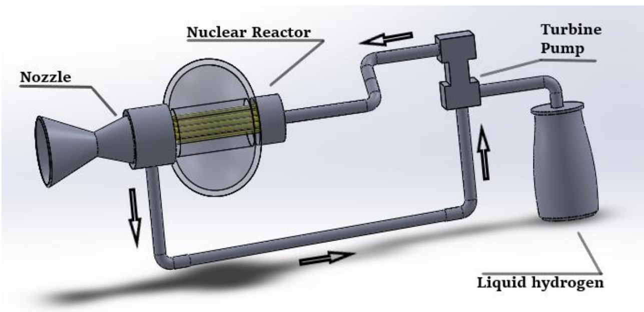


Fig 1: CAD model of NTP

II. SPECIFIC IMPULSE

Rocket propulsion is achieved by providing a thrust to the rocket which is obtained by the ejection of the propellant mass through the nozzle. The Specific impulse is considered as the important parameter in rocket propulsion which is nothing but the time integral of the thrust F(t) per unit weight 'w' of the propellant.

Whereas the total weight of the propellant is given as;

$$w = g\int_0^t m(\dot{t})dt$$

$\dot{m}(t)$ is the mass flow rate of the propellant and 'g' is acceleration due to gravity (9.8m/s^2)

Also,

$$I_{SP} = \frac{(F.t)}{W} = (F.t)/mg = V/g$$

The specific impulse helps to indicate the rocket efficiency and it also helps to compare between different rockets engines. It also signifies the sizing of the engine like if the value of specific impulse is known, the propellant mass which is used to provide the required thrust can be determined.

In 1903, A Russian school teacher formulated an equation to determine the equivalent velocity change of the rocket. This equation is known as Tsiolkovsky's rocket equation. [1]

$$\Delta V = V \log_e\left(\frac{Mp}{Mf} + 1\right)$$

Where, Mp is the mass of the propellant used and Mf is the final mass of the rocket.

The equivalent velocity change ΔV value is around 10 km/s for any rockets from earth to low earth orbit (LEO).

$$\frac{Mp}{Mi} = 1 - \exp(-\Delta V/I_{SP}.g)$$

The specific impulse of nuclear rockets is expected to be obtained as 1000sec whereas it is about 500sec for chemical rockets.

Following observations are made between specific impulse and the ratio of propellant mass to the final mass for both chemical rockets and Nuclear thermal rockets.

The specific impulse of nuclear rocket goes on increasing as mentioned in following table.

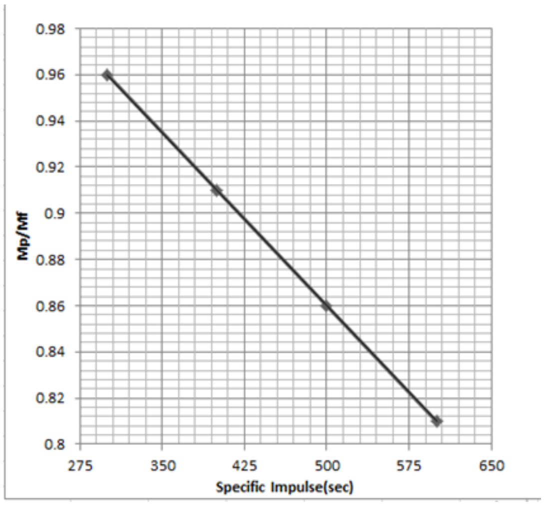


Fig 2: chemical rocket propulsion

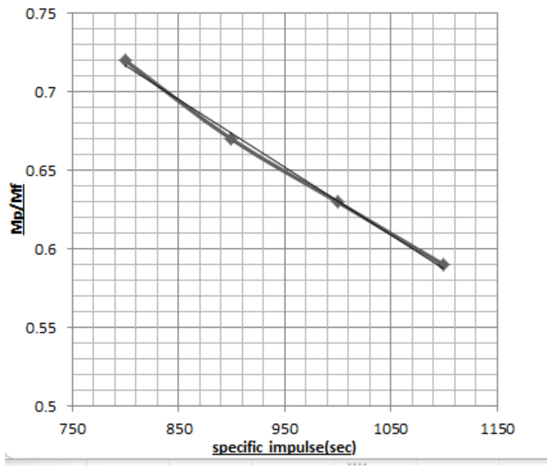


Fig 3: Nuclear Thermal propulsion

Table 1: specific impulse and $\frac{Mp}{Mf}$ value.

Nuclear rocket			Chemical rocket		
S. N	I_{SP} (sec)	$\frac{Mp}{Mf}$	S. N	I_{SP} (sec)	$\frac{Mp}{Mf}$
1	800	0.72	1	300	0.964
2	900	0.67	2	400	0.91
3	1000	0.63	3	500	0.86
4	1100	0.60	4	600	0.81

From the above graph, it shows that on increasing the specific impulse, mass of the propellant used is decreased. Thus, Nuclear rockets are more viable because of their high specific impulse value. The propulsive force is proportional to the specific impulse. The maximum propulsive force can be obtained if the specific impulse is maximum. The exit temperature of the propellant has direct relationship with specific impulse. Maximum the exhaust core temperature, maximum will be the specific impulse value as shown in given equation. [4]

$$I_{SP} = \frac{1}{g}\sqrt{\left(\frac{2\gamma}{\gamma-1}\frac{RT}{M}\right)\left(1-\frac{Pe}{Pc}\right)^{\frac{\gamma-1}{\gamma}}}$$

Where, γ is adiabatic constant i.e. C_p/C_v

R is gas constant. T is exhaust temperature of the propellant. M is molar mass of the propellant. Pe & Pc are the exit and chamber pressures respectively. Since the rocket is operated in space, the exit pressure is assumed to be zero. The nozzle should be designed in such a way which can withstand the exhaust temperature of propellant of about 3000K to 4000K.

III. SELECTION OF FUEL IN NUCLEAR ROCKET

The low molecular weight gas i.e. Hydrogen is selected as a propellant. The fuel is selected in such a way which has high uranium density in order to overcome the degradation of fissile materials and also to prevent it from various thermal attacks. Graphite based fuels and Cermet fuels are the potential fuels in Nuclear rocket propulsion engine. Also, we require the maximum exhaust temperature so carbide-based fuels are desirable than any other fuel because of its higher melting point. [10]

Carbide	M.P (K)	Thermal conductivity(W/m^2k)
(U, Zr)C	3350	10 – 30
(U, Zr, Nb)C	3800	20 -100
(U, Zr, Ta)C	3900	20 -100

The (U, Zr) C- graphite fuel is selected which is having 35% of carbide and 0.64g/cm^3 density of uranium which is sufficient enough to increase the reactivity of fissile materials to enhance the rocket performance. [9]. Thus, due to low melting point of cermet fuels (U, Zr) C-graphite fuel is chosen to conquer the maximum exhaust temperature.

The selection of the moderator has vital role to enable the use of LEU fuels in rocket propulsion. The moderator should have great neutronic performance, melting point and thermal conductivity. Thus, ZrH18 is selected as a baseline moderator.

IV. DEVELOPMENT OF LOW ENRICHED URANIUM

Research on development of LEU is being performed at NASA Marshal space flight center under the Space Capable Cryogenic Thermal Engine Program (SCCTE). [11]

The Through analysis in LEU-NTP shows that the LEU fuel can be applied in Nuclear Propulsion Technology and also the performance of LEU fuel will be similar to highly enriched uranium (HEU). A series of baseline cores have been proposed to exhibit the development needed for LEU Rocket Propulsion systems. The two reference cores are Space Capable Cryogenic Thermal engine (SCCTE) core and Superb Use of Low Enriched Uranium (SULEU) core. These cores are analyzed on the basis of NERVA/ROVER geometry using same materials and components.

The fuel being implemented is the primary difference between two reference Cores as SULEU uses (U, Zr) C-Graphite composite fuels whereas SCCTE uses enriched Tungsten -184 cermet fuel. The percentage of Uranium enrichment on both of the cores is 19.75% and is designed to operate with a nominal thrust of around 35klbf. [9] The Specific Impulse for SULEU is considered as 897.9 where for SCCTE is 894 as shown in Table 2.

Table 2: Specification of SULEU & SCCTE. [6]

Superb Use of Low Enriched Uranium (SULEU)		Space Capable Cryogenic Engine (SCCTE)	
Reactor System Mass		**Reactor System Mass**	
Fuel Mass (600 Elements) (kg)	800.1	Fuel Mass (151 Elements) (kg)	1029.8
Tie Tubes (427 Elements) (kg)	588.6	Tie Tubes (150 Elements) (kg)	700.4
Total Mass (Excluding Shield) (kg)	2498.0	Total Mass (Excluding Shield) (kg)	2557.6
Key Performance Parameters		**Key Performance Parameters**	
Nominal Isp (150:1 Nozzle)	897.9	Nominal Isp (150:1 Nozzle)	894
Nominal Thrust (kN)	155.7 (35k lbf)	Nominal Thrust (kN)	157.3 (35k lbf)
Whole Reactor Power(MW)	768.9	Whole Reactor Power(MW)	765.3
Fuel Temperature Max (K)	2850	Fuel Temperature Max (K)	2850
Fuel Details		**Fuel Details**	
Fuel Composition	(U,Zr)C	Fuel Composition	W-UO2-ThO2
Carbide Fraction (vol%)	35	^{184}W Enrichment (% atom)	98
Enrichment of ^{235}U (% atom)	19.75	Enrichment of ^{235}U (% atom)	13.13 to 19.75
Total ^{235}U (kg)	18.1	Total ^{235}U (kg)	45.9

V. MARS EXPLORATION & RADIATION PROTECTION

Obviously, Mars seems to be a distant goal for human exploration. Sending human to mars is challenging adventure but it is not impossible though. Perhaps after 2030 we will be seeing human into the mars because of advanced technology in rocket/spacecraft propulsion. Advancement in entry, descent, and landing (EDL) are required to land heavy payloads like human being & electric rovers in mars surface. Previous technology like mars science laboratory's sky crane was not capable to land payloads greater than 2 metric tons. But to land human being into mars surface, greater 40 metric tons payload is required. To reduce the weight of payloads, a rigid aeroshell concept has to be implemented i.e. ellipsled entry system. [8]

Astronauts will be exploring mars in pressurized rover which allows them to move beyond the landing site. Perhaps, Lunar electric rover (LER) would be the better rover for mars exploration too. The average speed of LER is 10km/h which can cover 60km in a day. To power this rover, lithium-ion batteries as well as regenerative fuel is provided. [8]

During the Mars exploration mission, Astronauts will be exposed to some harmful highly energetic cosmic rays. These space radiations may cause cancer and other harmful effects on crew's health. Due to this, radiation shielding has to be considered and designed. The materials containing hydrogen atom may be considered more effective at attenuating the protons & heavy ions. Thus, lightweight shielding material i.e. polythene is being studied.

Fig 4: Modelling

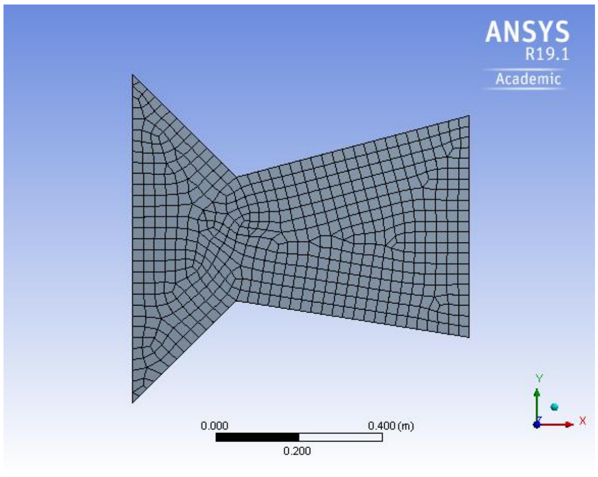


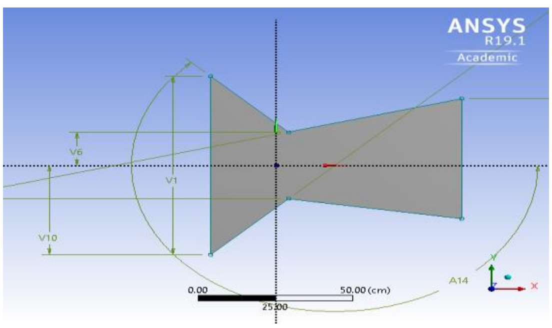
Fig 5: Meshing

Table 3: Design parameter of NTP Nozzle

Inlet diameter	80cm
Exit diamter	60cm
Throat diameter	30cm
Fluid	Liquid hydrogen
Specific heat capacity (Cp)	9772.2 J/kg.K
Density	70.85 kg/m^3
Thermal conductivity	0.10382 W/m.K
viscocity	1.332e-05 kg/m.s
Molecular weight	2.01594 kg/Kmol
Gauge total pressure	50e6 pascal
Supersonic initial gauge presssure	44.1e5 pascal
Temperature	3400K
Inlet velocity	8627.436 m/s

VI. DESIGN AND ANALYSIS OF NTP ROCKET NOZZLE

To examine the exhaust velocity of the rocket, a nozzle was designed which basically converts thermal energy obtained in the nuclear reactor chamber to kinetic energy. In this experiment, convergent divergent (C-D) nozzle was selected and its geometry was created using ANSYS WORKBENCH 19.1 as shown in figure 4.



After modelling, the meshing was done in the ANSYS software by automatic method. 50 number of divisions were taken in inlet and outlet side, where as 40 number of divisions were taken on side walls. The mesh was created of 630 nodes and 573 elements with linear element order. After meshing, setup is opened using ANSYS FLUENT. The Number of iterations is 2000 as shown in figure 6

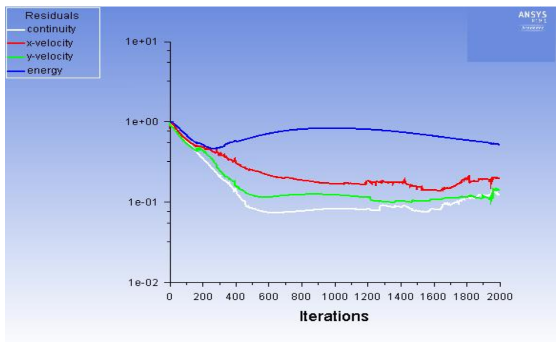


Fig 6: Convergence history

VII. RESULT AND DISCUSSIONS

The pressure becomes maximum at the inlet and goes on decreasing till the exit as shown in fig 7. The pressure at the exit is observed to be 0.3154 bar. The pressure decreases at the exit due to the occurrence of shock wave after the throat section.

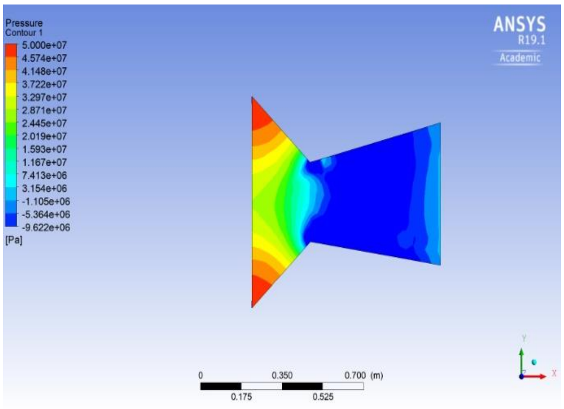


Fig 7: Pressure contour

The velocity at the inlet of the nozzle becomes minimum and goes on increasing till the exit. Since this is the nozzle of nuclear rocket so its velocity is higher than chemical rockets. The maximum velocity obtained at the exit is found out to be 9604m/s as shown in fig.8

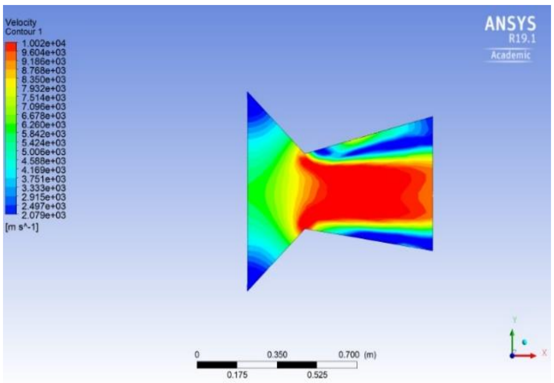


Fig 8: Velocity contour

Since the exit velocity from Nuclear thermal propulsion is approx. 9604m/s, thus the specific impulse obtained is 979sec.

VIII. CONCLUSION

Nuclear power has made an indisputable impact on the world for power generation. The need of high specific impulse in rocket propulsion is analyzed and the performance of nuclear rocket engine is studied. The higher specific impulse of the nuclear thermal rocket can reduce the mission duration than chemical rocket. Low enriched uranium (LEU) fuel helps in advancement of nuclear rocket propulsion technology. Thus, the mass of the propellant used can also be reduced by NTP technology.

In future, if the selection of fuel is done in such a way to achieve 900-1000 sec specific impulse then NTP would be better option for space exploration. To reduce the risks like leakage of gas, material degradation etc. induced by hydrogen propellant, some alternative source of propellant can also be used such as NH3, CH4, N2H2.

DECLARATION

Author had disclosed no conflicts of interest.

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