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Development of Kabila rocket: A radioisotope heated thermionic plasma rocket engine

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Abstract A new type of plasma rocket engine, the Kabila rocket, using a radioisotope heated thermionic heating chamber instead of a conventional combustion chamber or catalyst bed is introduced and it achieves specific impulses similar to the ones of conventional solid and bipropellant rockets. Curium-244 is chosen as a radioisotope heat source and a thermal reductive layer is also used to obtain precise thermionic emissions. The self-sufficiency principle is applied by simultaneously heating up the emitting material with the radioisotope decay heat and by powering the different valves of the plasma rocket engine with the same radioisotope decay heat using a radioisotope thermoelectric generator. This rocket engine is then benchmarked against a 1 N hydrazine thruster configuration operated on one of the Pleiades-HR-1 constellation spacecraft. A maximal specific impulse and power saving of respectively 529 s and 32% are achieved with helium as propellant. Its advantages are its power saving capability, high specific impulses and simultaneous ease of storage and restart. It can however be extremely voluminous and potentially hazardous. The Kabila rocket is found to bring great benefits to the existing spacecraft and further research should optimize its geometric characteristics and investigate the physical principals of its operation.

1. Introduction

Space is unarguably humanity's last frontier but its exploration has not been realized without any difficulties because its greatest obstacle is the lack of exploitable energy sources. Propellant and the efficiency of energy conversion systems that spacecraft must carry along with them limit their maximal range of operation. Chemical propulsion systems exhaust most of their propellant in order to reach stable orbits and thrust levels achieved by electric propulsion systems are limited by the production capacity of their power plants. It is thus rare to find chemical rockets with continuous burning time exceeding several minutes or electric propulsion systems with thrust levels greater than a few newtons.

Common methods to solve this problem have been used and this energetic scarcity problem has so far been tackled by improving the efficiency of propulsion systems and by opting for the most appropriate power plants in order to save energy through design optimization. Since further drastic

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improvements of the existing systems are unlikely to solve this energetic problem, new ways should therefore be sought.

Such way can be found in the use of radioisotopes. A new method to solve the problem of energetic scarcity in space will now be introduced. This method consists in replacing the catalyst bed and combustion chamber of mono and bipropellant thrusters with a radioisotope heated thermionic heating chamber. This method has the particularity of reaching high specific impulses without neither using solid nor bipropellant and also of rendering the rocket self-sufficient.

Radioisotope induced thermionic emissions were first introduced in DC discharge ion thrusters where the heater of a hollow cathode was then replaced by a radioisotope heat source. Its decay heat was subsequently used to power its electron discharge process hence resulting in significant power savings.[1,2]

Radioisotope rocket engines have already been developed, i.e., Poodle rockets, and investigated for interplanetary space travel.[3,4] However using a radioisotope heated thermionic chamber to heat up rocket propellant was never attempted before.

The purpose of this current work is to understand how this can be accomplished and what benefits such device would bring. The main features and characteristics of radioisotopes and rocket engines will first be outlined then suitable radioisotope heat sources will be selected before their performances may first be assessed and then finally discussed.

2. Engine operation principles

2.1. Rocket propulsion

The thrust generated by rocket engines is given by the following equation:

$$T = \dot{m} v_e \tag{1}$$

where $\dot{m}$ is the mass flow rate and v_e is the exhaust velocity. And it can be shown that thrust, rocket specific impulse and combustion chamber temperature are directly proportional:

$$T \propto I_{sp} \propto \sqrt{T_c} \tag{2}$$

where I_{sp} is the specific impulse and T_c is the combustion chamber's temperature. Rocket performances depend on the specific impulse and higher specific impulse can be achieved at higher combustion chamber temperature. Typical rocket combustion chamber temperatures obtained through chemical reaction range from 600 K to 4500 K with the corresponding specific impulse range of 180–450 s but plasmas could also be used to bring rocket propellants to such temperatures and could potentially reach better performances. Temperature is defined as the degree of particle agitation and plasma particles achieve much high temperatures than the ones of neutral gases because their ions and electrons are already separated hence can move much more freely. All matter eventually transits to a state of plasma as they are heated up because the degree of agitation of neutral particles is so high that the collision between them breaks the molecular bonds that link ions and electrons. Neutral gas heating is not the only way to generate plasma and gases can easily be ionized through collisions with an electron current of the right energy. Fig. 1 illustrates the schematic of this radioisotope heated plasma rocket engine, where l_e is the emitter material length, D_e is the emitter diameter and D_r is the radioisotope heater diameter. As previously explained, its geometry is similar to the one of conventional rockets at the exception of its combustion chamber. It is replaced by a radioisotope heated thermionic chamber where neutral gas is transformed into plasma through electron impact ionization. The chamber is composed of an emitter material, radioisotope heat source as well as of a radioisotope thermoelectric generator (RTG) and shielding layer which are respectively added to convert residual decay heat into electricity and attenuate the effect of hazardous radioisotope radiation. The operation of the components of this rocket will subsequently be explained and discussed.

2.2. Thermionic emission

Hollow cathodes thermionically emit electrons and these emissions are described by the Richardson–Dushman equation[5] and it can be shown that the thermionic electron current is directly proportional to the square of the emitter temperature.

$$J \propto T_e^2 \tag{3}$$

Temperatures exceeding 700 K are necessary to achieve useful thermionic emissions and a wide range of thermionic emission current densities are also possible.[6] Materials with low work functions, such as scandate, initiate thermionic emissions at lower temperature because less energy is required for free electrons.

2.3. Radioisotope

Radioisotope decays are exothermic and can induce radioisotope surface temperatures of the order of several hundreds

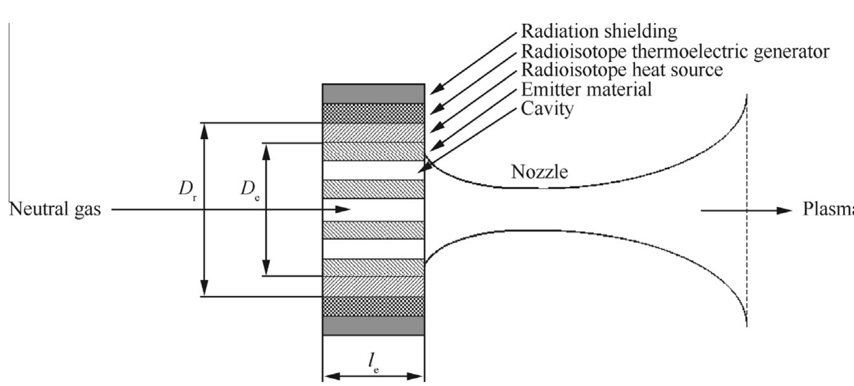

Fig. 1 Schematic of radioisotope heated plasma rocket engine.

and even thousands of degree.[6] Consequent thermionic electron emissions could be initiated through direct contact with such radioisotope heat sources and this method would have the advantage of not requiring any electricity because radioisotopes naturally decay. Radioisotopes currently used to power RTGs would be the most appropriate ones because they combine high decay heats, low gamma emissions and long half lives. Strontium-90 (^{90}Sr), Plutonium-238 (^{238}Pu) and Curium-244 (^{244}Cm) are such radioisotopes. Curium-244 is chosen for this application because its high specific heat and low density achieve the best geometric and mass performances among all the selected radioisotopes.[1] A thermal reductive layer could also be used to achieve more precise thermionic emission current densities.[1] Tables 1[7] and 2[8–10] give the characteristics of the selected radioisotopes, their shielding requirements and emission current densities.

2.4. Mass flow rate

The heating chamber of the plasma rocket transforms neutral gas into plasma similarly to the insert of a hollow cathode. Assuming that the neutral gas temperature inside a hollow cathode is equal to 2500 K,[6] the mass flow rate through a thermionic emitter is given by the cathode flow. Dividing a typical cathode flow by the insert orifice area gives the cavity mass flow rate density:

$$\dot{m}_{\mathrm{d}} = \dot{m}/A_{\mathrm{i}} \tag{4}$$

where A_{i} is the insert orifice area of a typical hollow cathode. The radioisotope thermionic heating chamber's diameter can be found by dividing the required mass flow rate by the mass flow rate density:

$$A_{\mathrm{hc}} = \dot{m}_{\mathrm{t}}/\dot{m}_{\mathrm{d}} \tag{5}$$

where $\dot{m}_{\mathrm{t}}$ is the required mass flow rate. These two equations can be used to size the heating chamber of the radioisotope heated thermionic plasma rocket engine.

2.5. Self-sufficiency principle

Additional benefits can be brought by applying the self-sufficiency principle.[1] It states that: "A self-sufficient subsystem is one that does not require others to fulfill its purpose within its system". Self-sufficient subsystems are very useful because they do not require any power inputs and can even output power into their system.

A subsystem can become self-sufficient in two different ways. It can either naturally fulfill its role hence does not require any power input from its system or can produce enough power through internal means to support its own operation.

This radioisotope heated thermionic plasma rocket qualifies as a self-sufficient subsystem by applying both approaches. It first naturally heats up its insert material up to emission temperatures then powers its hydraulic valves with electricity generated from its own decay heat using an RTG. During operation, the power generated by the RTG will exclusively be used to enable the rocket's operation however, unless used continuously, which will rarely be the case, the application of this principle has turned this rocket into an auxiliary power source that could be used to extend the range of operation of spacecraft's payload.

2.6. RTG

RTGs are devices that can convert heat into electricity. Static RTGs will be used in this study because their relative simplicity will keep them lightweight and compact. Although electric conversion efficiencies of 6.3% have already been reached,[11] a conservative value of 5% will be used instead.

Assuming that the radioisotope specific power is known, the electric power generated by an RTG can be directly calculated using the following equation:

$$P_{\mathrm{g}} = \eta_{\mathrm{c}} m_{\mathrm{r}} P_{\mathrm{spec}} \tag{6}$$

where η_{c} is the thermal conversion efficiency, m_{r} is the radioisotope mass and P_{spec} is the radioisotope specific power.

Table 1 Radioisotope characteristics and lead shielding required for 1 kW radioisotope sources for a target exposure of 10 mrem/h at 1 m.[7]

Radioisotope	Decay type	Decay energy (keV)	Half-life (y)	Compound form	Melting point (K)	Density (g/cm^3)	Watt per gram	Curies per watt	Pb shield required (m)
^{90}Sr	β^-	546	28.0	$SrTiO_2$	2313	4.6	0.22	148	6.0
^{238}Pu	α	5593	87.7	PuO_2	2673	10.0	0.39	30	0.1
^{244}Cm	α	5901	18.1	Cm_2O_2	2453	9.0	2.27	29	2.0

Note: 1 keV = 1.6×10^{-16} J; 1 mrem/h = 10^{-5} J/(kg · h).

Table 2 Thermionic emission current densities for Strontium-90, Plutonium-238 and Curium-244 using different insert materials.

Radioisotope	Surface temperature (K)[8–10]	Thermionic emission current (A/cm^2)		
		BaO-Scandate	Bao-w411*	LaB6*
^{90}Sr	1083	16	4.8×10^{-2}	9.4×10^{-6}
^{238}Pu	1343	132	1.9	4.3×10^{-3}
^{244}Cm	1273	88	0.81	1.0×10^{-3}

Note: *. In which, represents average value.

Values of radioisotope specific power have already been measured[12] but could be easily calculated[13] assuming that the radioisotope in question is predominantly an alpha or beta emitter. The radioisotope mass required to achieve a self-sufficient state can be found by first equating the target power generation to the thruster total power:

$$P_g = P_{tv} \tag{7}$$

The thruster total power is given by:

$$P_{tv} = \sum P_v \tag{8}$$

where P_v is the power required by a given valve to operate. The radioisotope power generated per unit mass can be obtained by using the following expression:

$$P_{sr} = P_g/m_r \tag{9}$$

The required radioisotope mass is given by:

$$M_{rr} = P_{tv}/P_{sr} \tag{10}$$

and the required radioisotope volume is equal to:

$$V_{rr} = M_{rr}/\rho_r \tag{11}$$

where ρ_r is the radioisotope density. It can be shown that the radioisotope volume is given by:

$$V = \pi l_e (D_r^2 - D_e^2)/4 \tag{12}$$

Equating Eqs. (11) and (12) yields the radioisotope diameter:

$$D_r = \sqrt{4M_{rr}/(\pi\rho_r l_e) + D_e^2} \tag{13}$$

Table 3 Data of 1 N hydrazine thruster configuration.[14]

Parameter	Value
Thrust (N)	1
I_{sp} (s)	220
Nominal mass flow rate (mg/s)	440

Table 4 NSTAR-TH15 data.

Parameter	Value
Insert diameter (cm)[6]	0.36
NSTAR cathode flow (sccm)[16]	3.7
Hollow cathode neutral gas temperature (K)[6]	2500
Emitter length (cm)[17]	6

Note: For xeon, 1 sccm = 0.0977 mg/s.

Table 6 Power related data.

Parameter	Value
Thruster-catalyst bed heater (W)[14]	6.4
Thruster-valve$_1$ 16 V DC (W)[14]	6.5
Thruster-valve$_2$ 28 V DC (W)[14]	9.5
Thruster- valve$_3$ 28 V DC (W)[14]	9.5
Thruster -total power (W)	31.9
Pleiades-HR-1 – power generation – EOL (W)[15]	1.5×10^3
Pleiades-HR-1 – mean power generation (W)[15]	850
Pleiades-HR-1 – instrument power requirement (W)[15]	400
Pleiades-HR-1 – number of 1 N hydrazine thrusters[15]	4

Note: EOL represents end of life.

3. Results

The performance of this radioisotope heated thermionic plasma rocket engine is assessed using the configuration of the hollow cathode of the 15th throttle of the NASA solar electric propulsion application readiness (NSTAR) thruster and the configuration of a 1 N hydrazine thruster developed by EADS (European Aeronautic Defense and Space Company) Astrium[14] which was operated on both spacecrafts of the Pleiades-HR-1 constellation.[15] Tables 3–6 give the data of the 1 N hydrazine thruster, hollow cathode of the NSTAR-TH15, noble gases used and the power configuration of the thruster and spacecraft. All the equations developed in this paper, the exhaust velocity of an ideal rocket and specific impulse equations were used with the data provided from Tables 3–6 to obtain the performances of the plasma rocket engine listed in Tables 7 and 8 and plotted in Figs. 2 and 3. Table 7 provides the mass flow rate and geometric performances achieved by different neutral gases operated at the combustion chamber temperature achieved by this plasma rocket engine, i.e., 2500 K.[6] Table 8 provides the power savings and mass requirements achieved by different neutral gases, which are then compared with the 1 N hydrazine benchmark thruster. The heating chamber diameter obtained in Table 7 is then plotted against the required thrust in Fig. 1 and the specific impulses achieved by the neutral gases for different gas exhaust velocities are illustrated in Fig. 2.

4. Discussion

4.1. Novel type of plasma rocket

It is demonstrated in this work that radioisotopes can be used in plasma rockets to thermionically ionize neutral gas and generate electricity.

Table 5 Noble gas data.

Parameter	Value				
	Helium	Neon	Argon	Krypton	Xenon
Universal gas constant (J/(K · mol))	8.314	8.314	8.314	8.314	8.314
Molar mass (g/mol)	4	20.18	39.95	83.80	131.29
Specific gas constant(J/(K · kg))	2075	412	208	99	63
Specific heat ratio (at 273 K)	1.63	1.642	1.667	1.689	1.666

Table 7 Mass flow rate performances.

Parameter	Value				
	Helium	Neon	Argon	Krypton	Xenon
Exhaust velocity (m/s)	5185	2295	1613	1103	890
I_{sp} (s)	529	234	164	112	91
Mass flow rate per orifice (mg/s)	1.1×10^{-2}	5.55×10^{-2}	0.11	0.231	0.361
Insert area (cm^2)	0.102	0.102	0.102	0.102	0.102
Mass flow rate density ($mg/(s \cdot cm^2)$)	0.108	0.545	1.08	2.27	3.55
Heating chamber area (cm^2)	4.07×10^3	8.07×10^2	4.07×10^2	1.94×10^2	1.24×10^2
Heating chamber diameter (cm)	72	32	22.8	15.7	12.6

Table 8 Power performances.

Parameter	Value				
	Helium	Neon	Argon	Krypton	Xenon
Selected radioisotope	Curium-244	Curium-244	Curium-244	Curium-244	Curium-244
RTG conversion efficiency (%)	5	5	5	5	5
Radioisotope specific power (W/kg)	2.27×10^{-3}	2.27×10^{-3}	2.27×10^{-3}	2.27×10^{-3}	2.27×10^{-3}
Radioisotope density (g/cm^3)	9	9	9	9	9
Required radioisotope mass (kg)	0.281	0.281	0.281	0.281	0.281
Required radioisotope diameter (cm)	72.03	32.15	22.92	15.94	12.83
Overall power saving (%)	9	9	9	9	9
Mean power saving (%)	15	15	15	15	15
Instruments power saving (%)	32	32	32	32	32

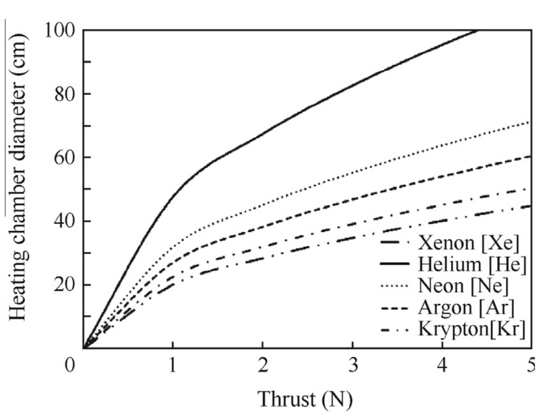

Fig. 2 Radioisotope heated thermionic heater chamber changing with thrust generated with different neutral gases.

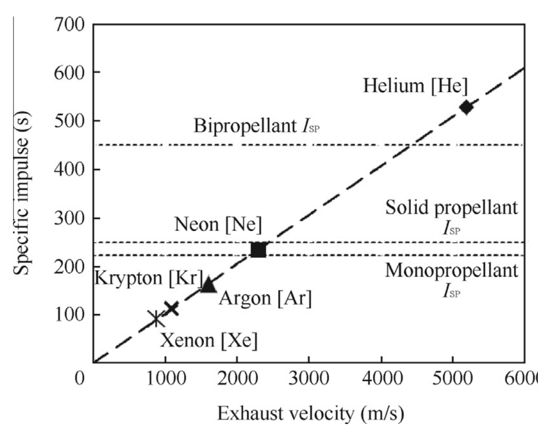

Fig. 3 Radioisotope heated thermionic plasma rocket engine specific impulse changing with exhaust velocity with different neutral gases.

Such combination of a radioisotope heat source and RTG in a plasma rocket is truly innovative and leads to the invention of a totally new type of plasma rocket which is named the "Kabila" rocket in honor of the late, Laurent-Désiré Kabila, hero and former president of the Democratic Republic of Congo.

4.2. Geometric consideration

The mass flow rate density has a great influence on the Kabila rocket and it should be reduced in order to achieve better geometric performances. This can be achieved in two different ways. First the discharge current cathode mass flow is calculated to generate a self-sustained discharge inside the emitter; however, thanks to the use of radioisotopes, such mechanism is not required anymore because heat is continuously supplied to sustain thermionic emissions. Neutral gas densities could therefore be increased without fearing that its temperature may decrease or that the thermionic emission may be interrupted. Second, the diameter of the insert material could be reduced to improve the geometric performance of the plasma rocket engine. The values used are based on a continuous operation of the NSTAR thruster hollow cathode over a period of several years. However, hydrazine thrusters are only required for attitude control and correction and will not be subject to the same requirements as the NSTAR thruster.

Much lower insert diameters are therefore expected on the Kabila rocket which would yield better geometric performances. A small quantity of radioisotope is required to power the valves of the rocket engine but a greater quantity would in fact be needed to quickly initiate thermionic emissions. The dimensions of the heating chamber nevertheless remain extremely high when compared with conventional hydrazine thrusters. A 400 N hydrazine thruster has approximately a 7 cm nozzle diameter[14] while the best performing radioisotope heated thermionic plasma rocket engine currently requires a heater chamber of 12 cm to produce the same mass flow conditions as a 1 N hydrazine thruster.

4.3. Power consideration

This rocket engine brings considerable energetic benefits by saving a not negligible amount of power which is equivalent to the sum of its valves' power. During operation it is self-sufficient and therefore does not require any power input from the spacecraft and can even supply power to the spacecraft when switched off. A single rocket engine can provide 32 W of electric power and each spacecraft of the Pleiades-HR-1 constellation operates four of them. This is equal to 128 W of additional power that could be used to supply up to 8.5% of the spacecraft total power generation, 15% of its mean power generation and even 32% of the power required by its instruments. Four additional instruments could thus have been supported thanks to this new rocket engine.

4.4. Comparative studies

The performance of the Kabila rocket exceeds the ones of conventional rocket engines but it is bulkier. It can achieve a wide range of specific impulses using different noble gases as propellant and the one achieved with helium exceeds those of traditional bipropellant rocket engines because helium has an extremely low molar mass. It however requires extremely larger emitter diameters to generate relatively small amount of thrust because of the same reason. A trade-off must therefore be found between achieving high specific impulses and reasonable amount of thrust. The use of this new plasma rocket engine will therefore be recommended for low thrust applications because the size of the heating chamber is the main obstacle of high thrust generation. Neon comes next in terms of ISP and achieves a specific impulse slightly lower than the one generated by solid propellant rockets.

4.5. Plasma rocket operation

The Kabila rocket is operated as follows. First a lifting mechanism puts the radioisotope heat source in contact with the emitter material then once it is heated up to thermionic temperatures, neutral gas is injected inside the emitter in order to be ionized. In its current state, the thermionic electrons produced by the plasma rocket do not possess enough energy to effectively ionize the neutral gas because of their relatively small mass and of the space charge effect which limits the thermionic electron current density. An electric field must therefore be introduced to accelerate thermionic electrons so that they may generate enough ions through neutral gas ionization because it is the collisions of the neutral atoms with these ions and the wall that will generate neutral gas temperatures exceeding 2500 K. The RTG should power this electric field and the required radioisotope diameter should be increased accordingly. A high temperature plasma is then produced inside the emitter and generates thrust by expanding inside the nozzle. Although the emitter material must be heated up similarly to the catalytic bed of monopropellant rocket engines, it uses a different exothermic process. Monopropellant rockets use chemical reactions to raise the gas temperature while radioisotope heated thermionic plasma rocket engines accomplish it through gas ionization. The radioisotope heat source needs to remain in contact with the emitter throughout the operation of the Kabila rocket which is switched on and off by disconnecting the radioisotope heat source similarly to radioisotope heated hollow cathodes.[1]

4.6. Advantages and disadvantages

The Kabila rocket has several advantages. First it can save a non-negligible amount of power, i.e., up to 32% of the power required by the instruments of one of the spacecraft of the Pleiades-HR-1 constellation. Second it can achieve very high specific impulse, exceeding those of solid and bipropellant rocket engines, i.e., 529 s. Its propellant is extremely abundant, i.e., helium, can be easily stored and it can also be restarted. Noble gases do not required special thermal conditioning as opposed to bipropellants and can therefore be stored for longer periods of time and at no additional energy cost. It also achieves specific impulses superior to the ones of bi and solid propellant rocket engines but as opposed to the latter, its combustion process can be initiated, interrupted or even operated in pulse modes as easily as mono and bipropellant rockets, adding to its precision and maneuverability.

The Kabila rocket would greatly benefit the existing spacecraft by extending their operation duration and range, lowering their operating cost and facilitating their routine maneuvers. The rocket can generate great power and fuel savings by, on the one hand acting as an independent power source and removing the need of the use of power-demanding cryogenic cooling systems to achieve specific impulses similar to or exceeding the ones of liquid propellant systems and on the other hand, generating far greater specific impulses. This would enable communication satellites, earth observation satellites and space probes to increase the extent of their payload and also their operating lives and ranges. This would result in a greater profit generation in the case of communication and commercial earth observation satellites and in more versatile and powerful applications for all types of earth observation satellites and space probes by for instance enabling the latter to operate in regions yet to be explored of deep space where lower solar density as well as ISPs of current space propulsion systems has so far hindered the operation of current space probes. The use of an inexpensive and abundant fuel such as helium will further reduce the operating cost of spacecraft. Communication satellites would therefore be able to generate further profit for their operators and running earth observation satellites as well as space probes' missions would become more affordable. This high ISP rocket can be more conveniently fired and restarted than equivalent liquid propellant systems. This would enable communication and earth observation satellites to periodically fire their thrusters to control their attitude as well as space probes to adjust their trajectories at a much lower fuel cost than it was previously possible.

The Kabila rocket also has disadvantages. Its heating chamber like most radioisotope heated hollow cathodes tends to be extremely voluminous and potentially hazardous. First the volume occupied by the heating chamber is currently extremely large but must also include an RTG module and lifting mechanism in addition to the emitting material and the radioisotope heat source. Finally radiations emitted by radioisotopes can be very hazardous and could either harm nearby operators or damage surrounding equipment.

These shortcomings should be tackled because they would increase spacecraft launch cost, limit the Kabila rocket's range of operation as well as render the launch and retirement of spacecraft potentially dangerous. Greater propulsion systems' dimensions necessarily mean greater launch mass which will directly affect the launch cost. With its current geometric performance, the Kabila rocket is restrained to a range of operation of only a few newtons. This range is only suited to low thrust applications such as earth observation attitude correction thrusters and a far greater range of operation of several hundred newtons would be required to service orbit transfer applications used by lower launch cost communication satellites and automated transfer vehicles. Without appropriate shielding, radioactive materials would either be dispersed in the atmosphere in the event of a failed launch or would create a cloud of radioactive space debris when spacecraft are retired causing potential damage to operational satellites in earth orbit and endangering the crew of manned space missions working in the International Space Station or other national manned space platforms.

5. Conclusions

A radioisotope heated thermionic plasma rocket engine, named the Kabila rocket, is successfully developed by using Curium-244, which is found to be the most suitable radioisotope because of its high specific heat and low density, to initiate the thermionic emissions that would through ionization increase the temperature of its propellant.

The self-sufficiency principle is applied by simultaneously heating up the emitting material with the radioisotope decay heat and by powering the different valves of the Kabila rocket with the same radioisotope decay heat using a radioisotope thermoelectric generator. This transforms the rocket into an auxiliary power source when it is not in operation.

The Kabila rocket is then benchmarked against a 1 N hydrazine thruster configuration operates on one of the Pleiades-HR-1 constellation spacecraft and a maximal specific impulse and power saving of 529 s and 32% are achieved with helium as propellant. Its advantages are its power saving capability, high specific impulses, as well as easy storage and restart. It can however be extremely voluminous and potentially hazardous.

The development of this engine has opened a new field of research and further work in this field ought to find ways to reduce the scale of the rocket engine by improving the mass flow density and also ought to aim at increasing the understanding of the physical principals governing its operation.

Further research in this field will focus on finding ways to reduce the scale of the rocket engine by improving the mass flow density and also be aimed at increasing the understanding of the physical principals of its operation.

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