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Preliminary Design of a Space Propulsion System Utilizing Stored Thermal Energy

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A space propulsion system utilizing stored thermal energy is analyzed with emphasis on the preliminary design of the heat exchanger and on the general performance of engines incorporating this concept. A high-energy propellant is developed by heat transfer from a phase-change material (Si) to a low-density gas (H_2) through a concentric cylinder heat exchanger. The phase-change material is heated by a radioisotope (PoGd) over a period of time which is significantly longer than the firing time. It is shown that such an engine can most efficiently operate just above and just below a propellant flow regime defined by gradual flow transition in the heat exchanger. Two designs are developed to illustrate each distinct operating scenario. A 10-N engine utilizing laminar flow is shown to be an attractive alternative to chemical and electric systems for the maneuvering of a vehicle in the vicinity of a larger space structure. A 450-N engine utilizing primarily turbulent heat transfer is shown to be an alternative to other systems for orbit transfer. Rapid cooling of the thermal storage material in the entrance region of the heat exchanger, which can be mitigated by varying the flow area or increasing the length of the engine, is a primary limitation on an engine utilizing thermal storage.

Nomenclature

c_p	=	specific heat
D_H	=	hydraulic diameter
I_{sp}	=	specific impulse
H_f	=	latent heat of fusion
H_v	=	heat of vaporization (propellant)
k	=	thermal conductivity
L	=	heat exchanger length
M	=	Mach number
m_{TES}	=	mass of thermal energy storage (TES) material
$\dot{m}$	=	propellant mass flow rate
Re	=	Reynolds number
T_m	=	melting point
t_b	=	firing time
$t_{1/2}$	=	radioisotope half-life
ΔT	=	change in temperature
η_{th}	=	thermal efficiency
ρ	=	density
τ	=	ratio of actual time fired to maximum possible time fired at a constant I_{sp}

Introduction

IN general, there are aspects of state-of-the-art space propulsion systems that present challenges to their respective uses. One is that oxygen or particulates in the exhaust of chemical systems may react with space structure surfaces. Another is that these systems tend to have either a low I_{sp} or a very low thrust. A third is that large power supplies, such as solar panels, on high-power electric systems limit rendezvous and eccentric orbital maneuvering. An alternative system which has the possibility of avoiding each of these limitations is a space propulsion system utilizing stored thermal energy.

The concept of utilizing stored thermal energy for spacecraft propulsion consists of transferring heat, generated initially by a small heat source, from a thermal energy storage (TES) material to a low-density propellant in such a way that an efficient thrust is developed. The concept was conceived by Hyder and Rose,[1] and was intended to be used primarily for small ΔV maneuvers. Initial concept feasibility studies were conducted by Lisano and Rose.[2]

The design and performance of two engines utilizing stored thermal energy for space propulsion is presented in order to illustrate the design feasibility and general performance of engines utilizing this propulsion concept. There are two principal objectives of the present research. The first objective is to quantify the characteristics of engines utilizing stored thermal energy, including the temperature distribution in the TES material, the performance of the engines to a first-order approximation, the general configuration and dimensions of the heat exchangers, and the requirements of engine materials. The second objective is to demonstrate the feasibility of integrating the engine into a realistic propulsion system and the operational utility of such a system. Determination of the limitations associated with engines employing this concept is, of course, also of primary concern.

Analysis

Preliminary studies have indicated certain advantages in various materials and configurations with respect to thermal storage propulsion applications. These studies have relied to a large degree on information compiled with respect to space power applications (e.g., Olszewski and Simon-Tov[3]). Results of the preliminary analysis include the selection of a heat source, thermal storage material, heat exchanger geometry, structural materials, and propellant.

Heat Source

The choice of heat source must be based on the consideration of factors such as the operating temperature range, the component masses, component reliability, and dependence, if any, on external power or control systems. Candidate heat sources include electric, nuclear (radioisotope), solar, and waste heat from an external source such as a space station. The electric source would consist of resistance heating elements located within the TES material, an external power source, and an external control system. The nuclear heat source

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would be a radioisotope, encapsulated in such a way that appreciable gas pressure does not develop. Since isotope heat generation would be continuous, a thermal management system would in all likelihood be necessary, unless surface radiation were sufficient to maintain an acceptable steady-state temperature in the TES material. Solar heating would be accomplished through the use of a small solar collector in a manner similar to that used, e.g., with solar dynamic power systems.[4] This would require an external control and actuator system as well. Waste heat is the energy generated by power systems which would normally be rejected to space through a radiator. Control devices would be required here also since only a portion of the rejected heat could be stored in the TES material.

The operating range for the heat exchanger portion of the propulsion system is assumed to be on the order of 700–2500 K, corresponding to the temperature at which hydrogen propellant produces an ideal specific impulse equivalent to that of a high-performance chemical system, and the temperature at which structural material limitations become of concern, respectively. The limiting factor for electric heat sources is the structural integrity of the heating elements. The use of radioisotopes would be limited by the thermal properties of the TES material, i.e., phase change temperature and boiling point. While no change in activity would occur, the volume change and a possible interaction with containment materials, leading to dispersion of the isotope or radiation leakage, would restrict operating temperatures. A solar heat source's operating temperature range would be determined by its proximity to the sun, the size of the collector, and the pointing accuracy of the collector. In general, the temperature of fluids carrying waste heat in a spacecraft is below the minimum acceptable temperature for effective operation of the heat exchanger.[5] A notable exception would be the heat rejection system associated with a nuclear reactor, especially one involving electric power generation, where the fluid temperature could reach 2000 K.[6] It is obvious, therefore, that operational temperature range is a factor in the section of isotopes, the size of a solar powered system, and the use of waste heat.

A preliminary comparison of subsystem masses indicates an advantage for the use of a radioisotope source. Less than 1 kg of some source materials and a similar amount of containment material will generate a few kilowatts of thermal power. The mass of an electric system lies primarily in the size of the power source, and is generally on the order of 2–6 kg/kW.[7] In the event that an external power source is available, e.g., power from a large space structure or even from a payload, the mass of the required power transfer mechanism would be on the order of that for the radioisotope source. The mass of a distribution system for waste heat would be determined by the proximity of the engine to the heat source, assumed to be a reactor, and by the transfer duct diameters required. Both of these are strongly dependent on the vehicle configuration and, in general, will cause the mass to be slightly greater than that of an electric system.[5] The mass of a solar collector and related equipment would be on the order of 20 kg/kW.[8]

Though radioisotope decay appears attractive as a heat source, it should be noted that its use introduces disadvantages not present with other methods. There is no throttling of the amount of heat generated, so that a heat rejection subsystem must be incorporated. The rate of recharging the TES material will decrease as the isotope decays. Radiation shielding, reentry hardening, and disposal plans for the engine at end-of-life are also concerns. Nonetheless, a radioisotope heat source is selected for the present study because of its high-power density, its independence from external power supplies, and the ease with which it may be distributed within the TES material and incorporated into various geometries.

The properties of several isotopes are listed in Table 1.[9–11] Isotopes with a high emission of beta or gamma radiation are not desirable because of the requisite shielding associated with them. If half-life is too short, significant changes in power output over the propulsion system's lifetime would be present. If half-life is too long, disposal of the engine at the end of its lifetime becomes an issue. The operating temperature range of the TES material utilized also places an important constraint on the selection of an isotope. PoGd is an attractive candidate for many missions because it has a high-power density and decays directly to lead. If the propulsion system were designed for, e.g., 3 yr of operation, the remaining level of activity should allow the isotope to be returned to Earth at any time after this. For long duration missions such as deep space probes, PuO_2 would be an attractive choice.

Table 1 Properties of high-energy isotopes

Isotope	Power density, W/g	Emission	T_M, K	$t_{1/2}$, y
^{238}Pu	0.56	α	911	87.0
$^{238}PuO_2$	0.40	α	2423	87.0
^{210}Po	141.3	α	527	0.378
$^{210}PoGd$	82.4	α	1930	0.378
$^{242}Cm_2O_3$	42.8	α/γ	2273	0.447
$^{90}SrO_2$	0.42	β	2730	28.0

Table 2 Properties of candidate TES materials

Material	ρ, kg/m^3	H_f, MJ/kg	c_p, kJ/kg K	k, W/mK	T_M, K
LiH	775	2.9	6.3	4.0	959
LiF-CaF	2097	0.753	1.8	4.1	896
Li	497	0.663	4.3	43.1	453
BeSi	3100	1.812	2.5	6.1	1363
Si	2330	1.884	0.7	23.7	1684

Thermal Energy Storage Material

The critical properties of any candidate TES material include specific heat, melting point, boiling point, coefficient of thermal expansion, density, latent heat, thermal conductivity, and material compatibility. In order to increase the specific impulse of a given mass flow, the temperature of the TES material should be as high as practical. For materials which rely on latent heat transmission, this would be a temperature slightly above the melting point of the TES material. For materials which utilize sensible heat transmission, the temperature limit is set either by the boiling point of the TES material or by the thermal limit of the structural material. However, operating temperature is not the only consideration. Specific heat capacity impacts both thermal storage capacity and recharge time. Thermal conductivity affects the manner in which a thermal load is distributed within the TES material. Latent heat capacity is critical for materials which utilize the energy associated with phase change. The coefficient of thermal expansion affects both the required containment mass and the propensity of the TES material to develop cracks as it cools. The properties of various candidate thermal storage materials are given in Table 2.[1,12] Though LiH is used quite often in space applications, Si and BeSi are attractive by virtue of their relatively high melting points. In particular, silicon appears to be attractive because of its combination of relatively high melting point, high thermal conductivity, and high latent heat of fusion.

Heat Exchanger Geometry (Internal Flow)

The flow path configuration is determined from considerations of heat transfer, physical dimensions of the heat exchanger, and material phase control. Material phase control refers to the ability to maintain the liquid phase of the propellant (if present) in contact with the flow path surfaces. Possible basic geometries include a single straight circular duct, a single helical circular duct, concentric cylindrical ducts, multiple circular ducts, and a spherical shell packed bed. A qualitative comparison of these geometries, as utilized in pro-

Table 3 Qualitative comparison of heat exchanger flow paths

Factor	Straight duct	Helical duct	Concentric cylinders	Packed bed	Multiple circular ducts
Heat removal	Fair	Fair	Good	Good	Good
Size	Poor	Good	Good	Good	Good
Phase control	Fair	Good	Good	Good	Fair
Thermal gradients	Good	Fair	Fair	Poor	Fair

pulsion-type applications, is shown in Table 3, taken from Ref. 13.

A straight duct configuration requires a long heat exchanger. A helical duct ensures that the liquid phase of the propellant remains in contact with the flow path surfaces, but is rather difficult to model properly.[13] Multiple circular ducts suffer from material phase control problems, as well as being susceptible to development of thermally isolated regions within the TES material due to temperature gradients. A packed bed heat exchanger offers a relatively large area for heat transfer, but, like the multiple circular duct geometry, suffers from problems with thermal gradients within the TES material. Overall, it appears that a concentric cylinder heat exchanger configuration represents a good choice because of its high surface area and ability to distribute thermal loads relatively uniformly. It should be noted that the heat exchanger configuration chosen in a NASA study[14] for quick, effective heat exchange utilized a concentric cylinder geometry.

Structural Materials

The selection of the structural materials, i.e., those materials in the engine which are not used for thermal storage, must be made on the basis of operating temperature range, density, thermal conductivity, machinability, and compatibility with the propellant. Such materials must be able to withstand system stresses caused by, e.g., propellant pressure, TES material thermal expansion, and thrust misalignment. Molybdenum and boron nitride coated graphite appear to be good candidate structural materials because of their high operating temperatures and molybdenum's compatibility with hydrogen, a likely propellant.

Propellant Selection

Since the concept of using stored thermal energy to transfer energy to a propellant is somewhat similar to nuclear-thermal and electrothermal propulsion concepts, hydrogen is an obvious propellant candidate. However, there may well be a tradeoff between molecular weight and thermal heat capacity. In any event, the propellant should be nonreactive with respect to external spacecraft surfaces and should be relatively easily storable. Possible propellants, in addition to H_2, include He, Ar, CO_2, NH_3, and N_2. Cryogenic materials will, of course, require special handling and storage facilities. However, an ongoing effort within NASA towards the commercial utilization of space should offer solutions to such problems within a realistic time frame.

Numerical Model

The aerothermodynamic principles which govern the design and operation of a space propulsion system which utilizes thermal energy storage can be roughly divided into three categories: 1) heat transfer, both within the TES material and to the propellant; 2) propellant flow properties; and 3) thrust generation. The scope and depth of the theory is determined by the goal of developing a preliminary design analysis of the performance and characteristics of such propulsion systems.

Heat Transfer

Of primary importance to the evaluation of the concept is a reasonable modeling of the temperature distribution within the heat exchanger, the rate and amount of heat transferred to the propellant flow, and the heat transfer, if any, to the surroundings. Conditions within the TES material must be known during steady-state periods of inactivity, during motor firing, and during recharging. While it is probably not necessary, at least in a preliminary design phase, to know the exact temperature at a given point, it is certainly necessary to be able to detect areas within the TES material which become thermally isolated from either the heat source or the propellant. In the present analysis, a two-dimensional finite difference iteration scheme is utilized to determine the temperature distribution within the TES material. A one-dimensional finite difference scheme involving influence coefficients (cf. Shapiro[15]) is utilized to determine the properties of the propellant flow through the heat exchanger.

Primary considerations associated with the use of TES materials which undergo phase change are the latent heat content and the possible change in material properties associated with the change-of-phase. The location of the phase change front as the thermal bed freezes or melts is also relevant. The method utilized here to model the phase change is the equivalent heat capacity method,[16] in which the phase change is assumed to occur over a range of temperatures in which the heat capacity is modified to include the latent heat of fusion.

A further constraint on the operation of a propulsion system using stored thermal energy is that of the efficiency of the heat transfer to the propellant. This efficiency must reflect the storage capacity of the TES material, the heat removed to the propellant, the operating temperature range of the heat exchanger, the length of time the engine is fired, and the variation in I_{sp} and thrust. Normally, thermal efficiency would be expressed simply as the ratio of heat removed by the propellant to heat stored in the TES material. However, in this instance a correction is made for the time of firing since the thermal efficiency will be low if the motor is fired for a time less than that necessary to remove all of the useful heat (i.e., heat stored above the TES material melting point, plus the heat of fusion of the material). Thus, the energy transferred to the propellant during the actual t_b is

$$\mathrm{ENERGY}_{\mathrm{transferred}} = t_b[\dot{m}(H_v + c_p\Delta T)]_{\mathrm{propellant}} \tag{1}$$

while the total energy available in the TES material is

$$\mathrm{ENERGY}_{\mathrm{available}} = m_{\mathrm{TES}}(H_f + c_p\Delta T)_{\mathrm{TES}} \tag{2}$$

If all the available energy were transferred ideally to the propellant, the firing time would be

$$t_{\mathrm{ideal}} = \frac{m_{\mathrm{TES}}(H_f + c_p\Delta T)_{\mathrm{TES}}}{[\dot{m}(H_v + c_p\Delta T)]_{\mathrm{propellant}}} \tag{3}$$

For actual t_b the available energy is modified by a factor τ, where

$$\tau \equiv (t_b/t_{\mathrm{ideal}}) \tag{4}$$

so that the expression for thermal efficiency becomes

$$\eta_{\mathrm{th}} = \frac{t_b[\dot{m}(H_v + c_p\Delta T)]_{\mathrm{propellant}}}{\tau m_{\mathrm{TES}}(H_f + c_p\Delta T)_{\mathrm{TES}}} \tag{5}$$

Propellant Flow Properties

In keeping with the preliminary design nature of the analysis, a number of simplifying assumptions are made with regard to the propellant flow calculations. The flow is assumed to be one dimensional, which implies the calculation of average, uniform properties at a given cross section. The gaseous phase is assumed to be thermally perfect. Inertial forces, such as acceleration during motor firing, and flow reactions, such as dissociation, are neglected. Changes in flow properties occur as a result of heat transfer, variations in cross-sectional area, and friction. As indicated previously, these changes are expressed in terms of influence coefficients.

If the propellant is initially in a liquid state, it will change phase somewhere in the heat exchanger. In the present model, once the propellant temperature reaches the critical value, its latent heat content is modified in discrete steps as the flow proceeds through the exchanger. Once the heat of vaporization is transferred, any further heat transfer once again raises the (now gaseous) propellant temperature.

Thrust Generation

Given the propellant properties at the exit of the heat exchanger, it is possible to determine the thrust characteristics and mission capabilities of a specified design. The nozzle is assumed to be converging-diverging, with a fixed expansion ratio and a fixed ratio of throat area to heat exchanger exit area. Flow in the nozzle is assumed to be calorically perfect, one-dimensional, and isentropic. The use of a nozzle efficiency factor is used to account for nonisentropic effects.

General Performance Characteristics

The overall engine performance is, of course, determined by the heat transfer to the propellant in the heat exchanger. Before proceeding to specific engine configurations, it is possible to discern several performance characteristics which appear to be generic to this type of propulsion system. Once a TES material and a propellant are selected, the primary design parameters for the heat exchanger are the length-to-hydraulic diameter ratio L/D_H and the flow Reynolds number, defined in terms of propellant flow rate and hydraulic diameter. In all of the calculations described below, certain common characteristics are shared by each configuration. Thermal control is accomplished passively through layered insulation. The insulation layers are corrugated ZrO_2, evacuated Si_2O_3, and evacuated layers of Al_2O_3, a combination which would allow for the radiation of an amount of energy equal to that being generated by the heat source when the outer TES material reaches a given maximum temperature. The thicknesses of the individual layers would vary, however, with each configuration. PoGd is the radioisotope heat source, and the isotope is assumed to be in its fourth half-life during operation. Hydrogen is the propellant, and Si is the thermal storage material.

Figure 1 illustrates the variation of I_{sp} with Re for several values of L/D_H ratio. Note that there is an operating region where I_{sp} decreases significantly with overall system Reynolds number. It appears that optimum system operation quite naturally divides into two distinct Reynolds number regimes. It turns out that low I_{sp} operation corresponds to the transition between laminar and turbulent flow in the exchanger, or around $Re = 8 \times 10^3$ (cf. Wilson and Medwell[17]), a result which could be anticipated from the behavior of the friction factor (and, through the Reynolds analogy, the heat transfer coefficient) in the transition region. The performance degradation is more severe and extends over a wider operating range of Reynolds number as the L/D_H ratio decreases. Again, this is not unexpected since lower L/D_H ratio implies either a shorter heat exchanger or a correspondingly lower value of heat transfer coefficient. Figure 2 illustrates thrust vs operating Reynolds number curves for the same L/D_H ratios as in Fig. 1. Comparing the two figures, it appears that the preferred operating regimes for this type of propulsion system are a low thrust, high I_{sp} regime with laminar flow throughout the heat exchanger and a moderate thrust, slightly lower I_{sp} regime with turbulent flow through the exchanger.

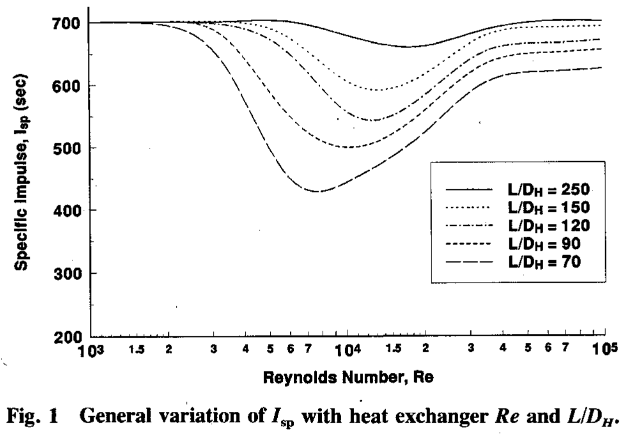

Fig. 1 General variation of I_{sp} with heat exchanger Re and L/D_H.

Figure 3 illustrates the relationship between thrust-to-weight ratio and operating Reynolds number for a system with a heat exchanger L/D_H ratio of 150. Thus, while the geometry is held fixed, the propellant pressure, and therefore, the propellant flow rate, is varied. The behavior indicated is not unexpected: the resulting thrust is a tradeoff between mass flow and heat transfer effects. The heat transfer effects are illustrated in Fig. 4, which shows the behavior of the thermal efficiency, as defined by Eq. (5), for the same operating system as shown in Fig. 3. It is obvious that a tradeoff between the thrust-to-weight ratio and thermal efficiency will be a primary design consideration for TES-type propulsion systems.

As indicated previously, the overall efficiency of the system is also a function of the firing time. In fact, the firing time is (next to the properties of the TES material) the single greatest constraint on the entire thermal storage concept. A principal reason is the time-wise development of large axial (flow direction) thermal gradients in the TES material, as shown in Fig. 5 for a heat exchanger with a ratio $L/D_H = 150$. These gradients develop because of the relatively higher temperature differences between the TES material and the propellant in the entrance region of the heat exchanger. The net effect of these large thermal gradients in the entrance region is to reduce the effective length of the heat exchanger. Such gradients become even more significant as operating Reynolds number increases due to the correspondingly higher heat transfer rates. What is happening is that the high initial heat transfer rate in the entrance region of the exchanger is effectively penalizing the system performance at a later time. As the firing time progresses, cooling of the TES material eventually tends to reduce the attainable propellant temperature, placing a constraint on the overall time for which a given constant specific impulse can be maintained. It is possible to reduce these TES material gradients by varying the area of the flow path annuli so that the heat transfer becomes more uniform along the axis of the exchanger. This allows the stored thermal energy to be more uniformly discharged over the desired firing time. A 20% increase in the flow path area at the entrance to the exchanger compared to that at the exit provides a significant decrease in the axial direction thermal gradient within the TES material, without a significant increase in heat exchanger mass. In fact, such an area variation theoretically allows a 50% increase in the firing time of the engine, as determined by the time a given flow rate could be maintained within a few degrees of a given, high-exit temperature. Though the 20% figure is not a true optimum, it does represent a tradeoff between heat transfer considerations and considerations of heat exchanger entrance geometry, and is in keeping with the preliminary nature of the present analysis.

Finally, the system performance obviously depends on the Mach number existing at the exit of the heat exchanger. Suppose that the heat exchanger length is fixed so that variations in L/D_H ratio occur only as a result of variations in the propellant flow path annuli dimensions. The overall diameter of the heat exchanger could then be maintained roughly constant by the simultaneous variation of the number of annuli, the height or thickness of a given annulus, and the thickness of the TES material between the flow path annuli. If the operating Reynolds number, and thus the exit Mach number, is

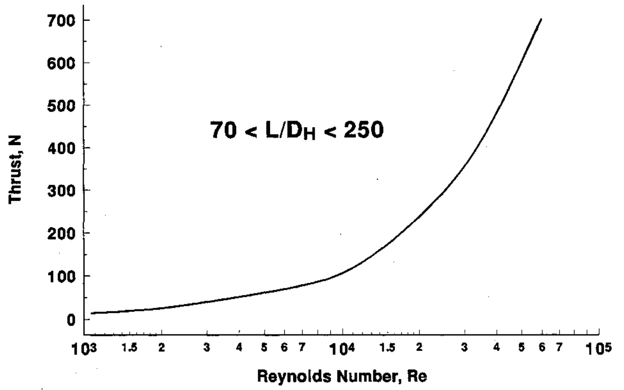

Fig. 2 General variation of thrust with heat exchanger *Re* and L/D_H.

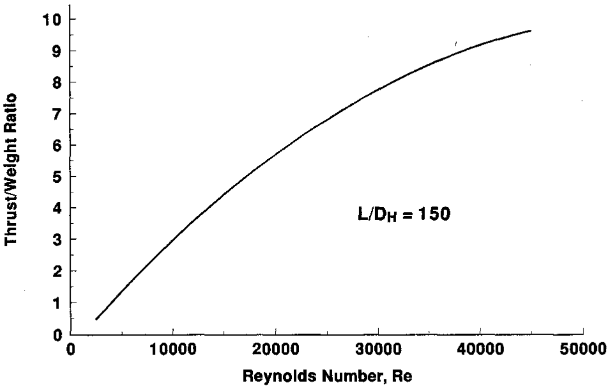

Fig. 3 Variation of thrust-to-weight ratio with heat exchanger *Re* for L/D_H = 150.

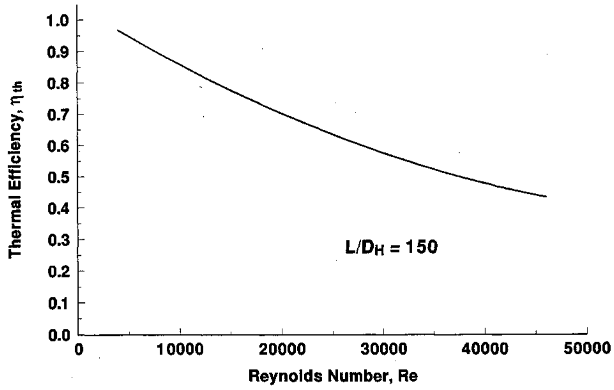

Fig. 4 Variation of thermal efficiency with heat exchanger *Re* for L/D_H = 150.

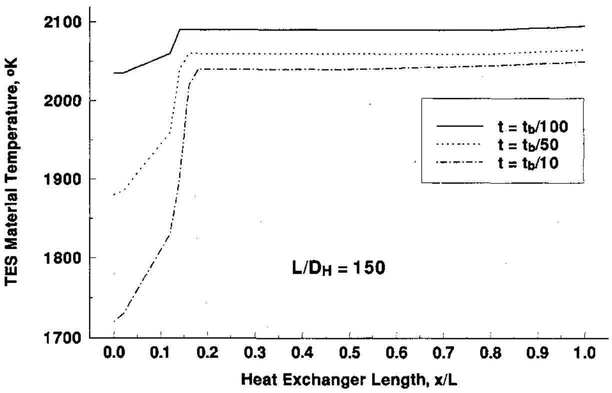

Fig. 5 Time-wise development of TES material flow-direction thermal gradients.

then varied by changing the operating pressure level (flow rate), one finds that 1) I_{sp} generally optimizes in the range M_{exit} = 0.05–0.20, depending on the L/D_H ratio, with the higher Mach numbers corresponding to the higher L/D_H ratios; 2) that lower values of M_{exit} are preferable for laminar flow heat exchangers, while higher values of M_{exit} are preferable for turbulent flow exchangers; and 3) that a larger number of annuli, with correspondingly smaller flow path and TES material thicknesses, is preferable from the standpoint of maximizing I_{sp}, no matter what the value of M_{exit}. The latter result comes from the higher L/D_H ratio.

Specific Configurations

Recall that the preferred operating regimes for propulsion systems utilizing stored thermal energy are a low thrust, high I_{sp} regime with laminar flow throughout the heat exchanger and a moderate thrust, slightly lower I_{sp} regime with turbulent flow throughout the exchanger. With this in mind, two specific configurations have been analyzed in some detail; each is designed specifically for one or the other of the thrust regimes identified above. Each is intended to be a "useful" design, i.e., meet basic performance criteria and utilize existing materials.

As in the section on general performance characteristics, Si is used as the TES material. The conductivity of the TES material is enhanced by the inclusion of a number of radial spars which extend the length of the heat exchanger. Inclusion of such spars is a tradeoff between heat distribution and energy storage capacity. The outer insulation for the heat exchanger is selected such that it allows the radiation of an amount of energy equal to that being generated by the heat source when the outer TES material annulus reaches a temperature of 2100 K. The heat source is the radioisotope PoGd, assumed to be in its fourth half-life; the peak power generation is therefore about 8.75 kW/kg. Hydrogen is selected as the propellant for both configurations because of its performance advantages. It is cryogenically stored and would be forced into the heat exchanger by means of a bladder controlled by high-pressure He. A pressure regulator upstream of the heat exchanger would be used to regulate the flow into the exchanger. The use of hydrogen propellant assumes the development of long-term cryogenic storage. The same engines will function, albeit at lower performance levels, if more storable propellants such as NH_3 are used.

10-N Thruster Configuration

The preliminary design of a 10-N engine utilizing stored thermal energy has been undertaken, given the following constraints: it is to be defined by its capacity to fire rapidly in variable impulse sequences without having a specific ΔV capability. Its general operating parameters are defined by the laminar region of the performance curves above. Because of this, no variation in flow area is introduced. A geometry generating a Mach number of 0.1 at the exit of the exchanger is utilized. The selection of annular thicknesses of 0.003 m for both the TES and flow path annuli yields a L/D_H ratio of 16.7. Five heat exchanger flow paths operating at a nominal pressure of 0.15 atm generate the requisite 10-N thrust.

Figure 6 illustrates the heat exchanger and general engine configuration, while the general characteristics of the engine are given in Table 4. The variation of thrust and I_{sp} of the engine is illustrated in Fig. 7. Note that the nominal thrust level can be maintained for several minutes, though the maximum attainable I_{sp} can be maintained for only about a minute. "Off-design" performance, i.e., performance at nominal pressures other than 0.15 atm, is shown in Fig. 8. At 4-atm pressure, a thrust of 300 N is possible, though only for a very short time. Obviously, the production of higher thrusts is critically dependent on the state of both fuel and thermal storage capacity at the time such thrusts are required.

Table 4 Characteristics of the 10-N engine

Nominal thrust	10 N
Maximum specific impulse	790 s
Nominal operating pressure	15.2 kPa
Mass	2.95 kg
Power generated	200 W
Thermal efficiency, design	0.95

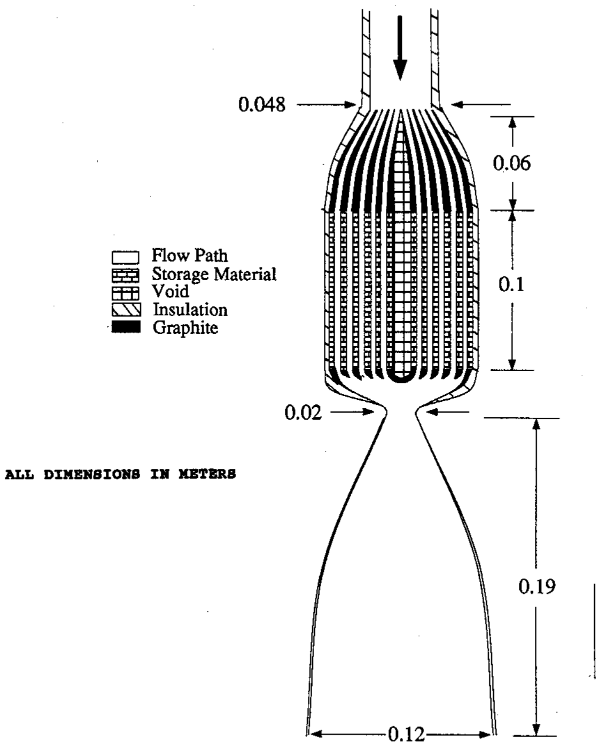

Fig. 6 10-N Engine configuration.

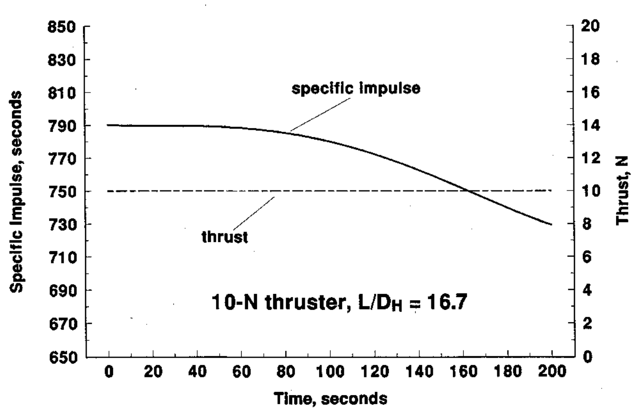

Fig. 7 Time-dependent performance characteristics of the 10-N engine.

450-N Thruster Configuration

The preliminary design of an engine utilizing stored thermal energy and developing a nominal thrust of 450 N has been undertaken to evaluate the performance of such an engine operating at moderate thrust levels. The system will operate at Reynolds number levels which ensure turbulent flow in the heat exchanger. Somewhat arbitrarily, an exit Mach number of 0.15 is imposed on flow leaving the heat exchanger. A converging flow path geometry with a convergence ratio of 5:4 (25% larger in the heat exchanger entrance region) is selected, with nominal flow path thicknesses of 0.004 m, giving a L/D_H ratio of 112.5. The TES material annular thicknesses vary from 0.003 m along the inner and outer radii to 0.005 m everywhere else. Four flow paths operating at a nominal pressure of 5 atm produce the requisite thrust at an I_{sp} of 670 s.

Table 5 Characteristics of the 450-N engine

Nominal thrust	450 N
Maximum specific impulse	670 s
Nominal operating pressure	506.6 kPa
Mass	13.0 kg
Power generated	1000 W
Thermal efficiency, design	0.90

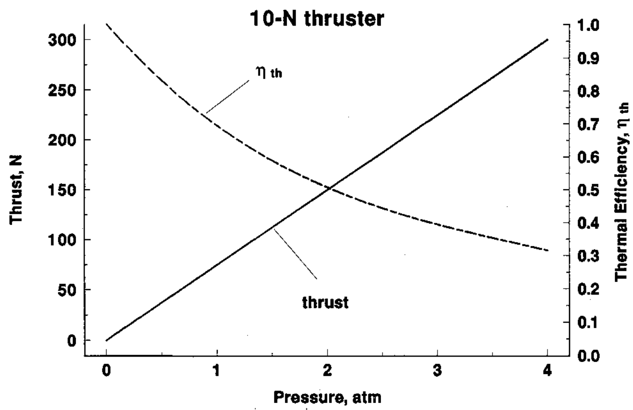

Fig. 8 Off-design performance characteristics of the 10-N engine.

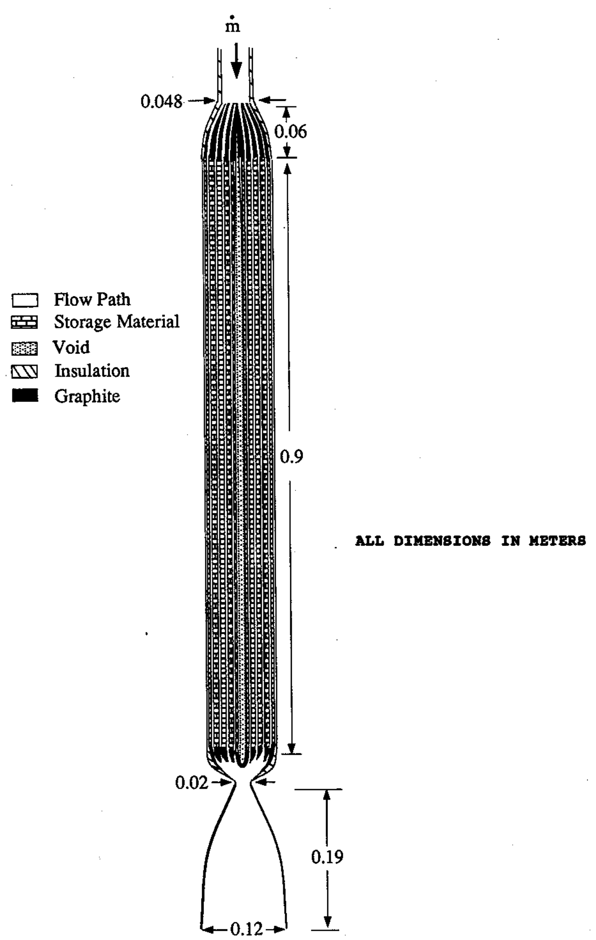

Fig. 9 450-N Engine configuration.

The general characteristics of the engine are given in Table 5, and the engine configuration is illustrated in Fig. 9. Figures 10 and 11 illustrate the nominal thrust and I_{sp} performance characteristics and the "off-design" performance, respectively, of the 450-N engine.

As with the 10-N thruster, the thrust is relatively constant with time. However, as indicated in Fig. 10, the time of useful operation is much shorter than that of the lower thrust con-

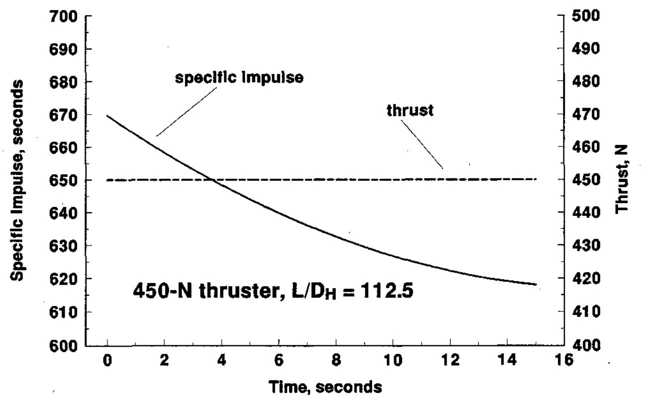

Fig. 10 Time-dependent performance characteristics of the 450-N engine.

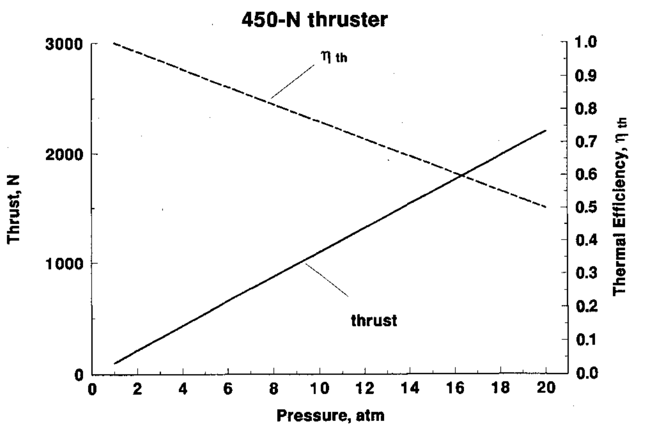

Fig. 11 Off-design performance characteristics of the 450-N engine.

figuration. As with the smaller engine, a large range in thrust appears to be possible. Note that both engines are quite efficient at thrust levels below their design operating points. Together, the 10-N and the 450-N engines generally represent the useful operating ranges of propulsion systems utilizing stored thermal energy.

Conclusions

The feasibility of integrating thermal storage into a space propulsion system has been demonstrated. Specifically, designs for both a 10-N and 450-N engine have been developed using first-order optimizations of materials, configurations, and operating conditions. While a radioisotope was selected as the heat source for this analysis, other heat sources are not precluded. Similarly, though hydrogen was selected as the propellant for this investigation, the use of indigenous propellants is also possible.

There are generally three applications of low-thrust engines: 1) large total impulse maneuvers over a long period of time, 2) attitude control, and 3) primary maneuvering of small vehicles. It is in the third area that the propulsion system utilizing stored thermal energy appears to be most attractive. Its larger thrust would allow more timely maneuvers than electric systems, and its I_{sp} and benign propellant present advantages over chemical systems. A system utilizing thermal energy storage is also attractive for attitude control if a range of impulses is required, e.g., that provided by multiple electric or chemical thrusters on a large space vehicle or platform. Thus, small engines utilizing stored thermal energy may well be a viable and attractive alternative to other systems for low-thrust operations.

Operations which might involve the engines in the 450-N class are primarily associated with orbital maneuvering. The TES-type propulsion system is primarily limited by the firing time of the engine, which cannot be increased significantly without a significant decrease in the thrust-to-weight ratio. The total impulse capacity of the engine can, of course, be increased by increasing the size of the heat source. The relative advantage of using a propulsion system utilizing thermal energy storage would then be determined by whether a multiple burn transfer would be more efficient, because of the thermal storage engine's higher I_{sp}, than single burns associated with chemical systems. That analysis is beyond the scope of this article, but there may be advantages to using the 450-N engine for large geocentric transfers, e.g., LEO to GEO, in terms of mass and time with respect to chemical and electric systems, respectively.

The primary limitations on a propulsion system utilizing stored thermal energy include the recharging time between firings; the rate at which the entrance region of the heat exchanger cools; the thrust to weight ratio, as determined by the decrease in thermal efficiency at high operating pressures; and a region of low performance associated with flow transition within the heat exchanger. The primary implication of these limitations is that propulsion systems utilizing stored thermal energy cannot be used efficiently for high thrust or high impulsive ΔV maneuvers. However, it has been demonstrated that these limitations do not preclude the development of useful engines for other applications.

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