X-736-67-388

NASA TM X-55884

DESIGN AND PERFORMANCE OF A THERMAL STORAGE RESISTOJET

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Abstract

A thermal storage resistojet has been designed and tested for possible satellite applications. The resistojet was built for NASA/GSFC by the General Electric Company.

An electrical resistance element is used to heat ammonia propellant to better than 1800°F giving specific impulse values of 230 seconds at 20 mlb thrust. Thirty watts is the power requirement to attain this temperature.

The mechanical design is based on a modular concept incorporating three (3) modules: heating element, thruster body and heat shield package. Hardware interchangeability, integrity and ease in design alterations for performance changes are assets derived from a modular approach.

The configuration of the thruster is that of a flange mounted cylinder two inches in diameter and 5 inches long. Performance testing for the resistance jet is done at both steady state and transient conditions. Pulse mode tests are conducted to simulate satellite applications and enable the engineer to predict how the thruster will function under any specific situation. Test data allows features including thrust vs. I_{sp} and heat shield effectiveness to be readily evaluated.

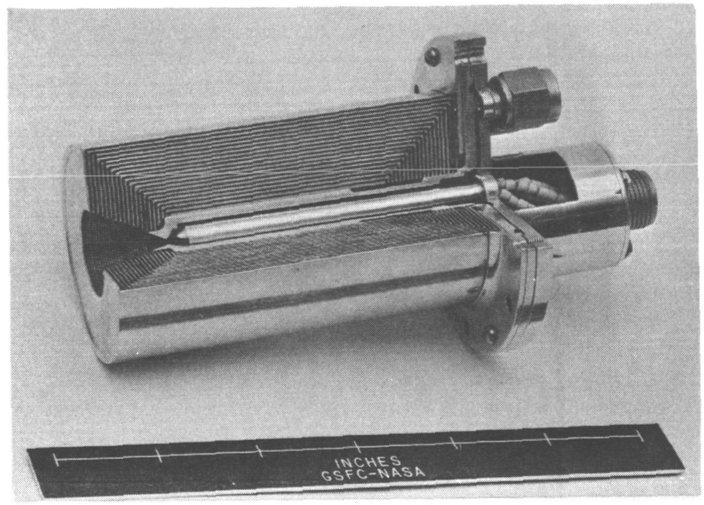
Environmental tests including vibration, acceleration and thermal vacuum along with the performance tests have shown the resistojet thruster design to be feasible for low thrust satellite applications.

Introduction

As spacecraft become larger with longer and more complex missions, the need for a continuing development of advanced, low thrust level, rocket engines to provide a means of controlling the spacecraft attitude and/or orbit throughout its operational lifetime becomes essential. Cold gas jets impose severe mission restrictions because of their low specific impulse and high pressure storage requirements. To improve mission capability, considerable effort has been devoted to developing systems such as ion and colloid engines, resistance jets, chemical rockets (both mono and bi-propellant), subliming solids, etc. One particular thruster, a thermal storage resistance jet, is described in this paper.

The design discussed in this paper is for a single nozzle thermal storage resistance jet thruster. This unit is referred to as a model TSK-2000-1P thruster. It was developed and built by the General Electric Company under contract to Goddard Space Flight Center.

Figure 1 shows a sectional model of the thermal storage resistance jet. The external envelope of the thruster is about 2 inches in diameter and 5 inches long with a circular mounting flange. The thruster package weighs approximately 1.7 lbs. The thruster assembly consists of a nozzle and flow annulus through which the propellant passes, a calrod type heater subassembly, and heat shielding fabricated from layers of thin metallic foil separated by minimum contact support wires.



SECTION - RESISTOJET
FIGURE 1

Characteristics of the thermal storage resistojet which make it attractive for station keeping and attitude control are:

1) The design flexibility which allows the trade-off of electrical power with propellant consumption
2) High specific impulse (I_{sp}) at low peak electrical power
3) Low pressure propellant storage and feed requirements (using ammonia as the propellant).

The principle of operation of the resistojet is that the propellant is heated by contact with a hot metallic surface over which it passes in the thruster annulus. The heat addition increases the system's specific impulse (reduces gas consumption). The resistance wire heater package which is installed in the center of the body provides the heat source. The heater element is designed to raise the gas temperature to 2000°F. It has sufficient heat capacity to allow the core temperature to remain almost constant for short propellant pulses. To minimize the required power and maintain high thermal efficiency, the thermal storage resistojet employs unique thermal insulation techniques. The design, fabrication and performance of the resistojet is described in detail in separate sections of this paper.

Ammonia gas was selected as the propellant for the resistance jet mainly because of its high total impulse/weight ratio. The properties of ammonia which make it attractive for this type of application are:

1) Can be stored as a liquid at comparatively low pressures
2) Has a low molecular weight (high I_{sp})
3) Dissociates at temperatures within the capability of the resistance jet (again improving I_{sp}).

Design and Fabrication

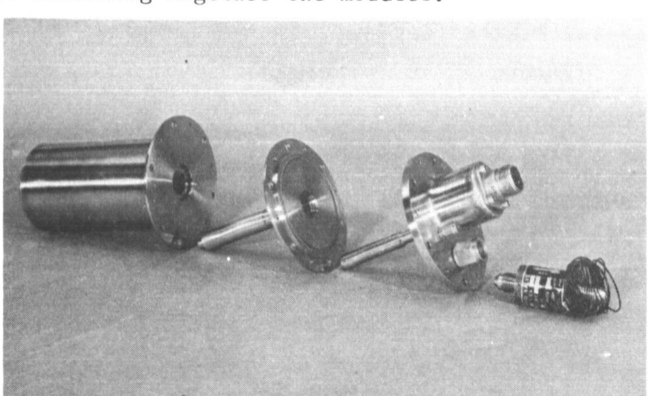
Performance requirements set forth for this thruster design were selected as typical for possible satellite attitude control and station keeping requirements and to advance the state of the art in resistance jet technology. The design requirements established are shown in Table 1.

Thrust Level	0.020 lb.
Chamber Pressure (P_c)	1.5 atm.
Core Temperature (T_c)	2000°F
Power Input Max. (No flow, T_c=2000°F)	30 watts
Propellant	Gaseous Ammonia
Minimum Allowable Impulse Bit	24×10^{-4} lb-sec
Maximum Reduction in I_{sp} for Three Minute Pulse	20%
Nozzle Alignment: Normal to Nozzle Thrust Plane to be Held to Within 1°	
Weight	1.75 lb.

TABLE 1

Guidelines followed in the mechanical design were simplicity, use of modular techniques, and structural integrity through the selection of materials with proven fabrication methods.

Three (3) Modular components make up the complete thruster. These modules consist of a heat shield package, a thruster body and an electrical heater unit. Figure 2 illustrates the modular breakdown. A prime advantage to the modular concept is that it permits interchangeability of the three precision made modules. Reliability and integrity is retained even with repeated assembly and disassembly through the use of bolt screw techniques for fastening together the modules.



MODULAR COMPONENTS - RESISTOJET
FIGURE 2

The merits of using modular techniques has been demonstrated in many other types of space flight hardware and is a significant advantage in thermal storage resistance jets.

The modular approach allows for readily altering performance characteristics without affecting design interfaces or envelope. This feature eliminates the need for major redesign effort and extensive requalification programs. When damage is incurred by a thruster, moreover, repair is accommodated by simple replacement of the damaged module and rarely would result in total destruction of a thruster.

Referring back to Figure 1, the structural design is illustrated. When the modules are assembled the unit, becomes one integral unit and when subjected to vibrational environments acts essentially as one mass, hard mounted at its base. The long slender heater element when mounted into the thruster body is supported at both ends and is in a state of axial compression. Likewise the thruster body is supported in the heat shield package, and the three modules are fixed together at the flanged surfaces.

The approach taken was somewhat conservative. This is demonstrated by the fact that each module is independently capable of withstanding severe vibration loads. A disadvantage to the structural design established is that there is a significant heat loss to the mounting flange since the heater element is welded directly to a flange. Thermal isolation of the heater element from the mounting flange would save an estimated ten (10) watts at 2000°F core temperature. Design modifications to provide for this savings are discussed later in this paper.

A detailed discussion of the design and fabrication of the TSK-2000-1P thruster is best accomplished by separate descriptions of the three modular components: the heat shield package, thruster body, and heater element.

The purpose of the heat shield package is to minimize the conduction and radiation heat loss by the thruster and thereby lessen the power input required to maintain the 2000°F temperature in the core. Considerable work has been done to optimize the number and type of shields. The desirable characteristics of the shielding material are low emmissivity, poor heat conductivity, light weight, non-reactive with the propellant (notably ammonia), and a minimum of fabrication problems. Nichrome V appears to be the best compromise material at this time.

Reference (1) presents an analytical technique used to estimate the number of shield layers required for a given operating heater temperature. The basic procedure followed is presented here.

To simplify the analysis the following assumptions were made:

1) The emmissivities of all layers of heat shielding are equal to an effective value (empirically determined)
2) The ratio of areas of adjacent layers of heat shielding is nearly the same throughout the shields.

With these assumptions an expression for the heat radiated from the thruster body at temperature T_1, with m layers of heat shielding, to free space was found to be

$$q = \frac{\sigma E T_1^4 \; [A_1+(m-1)K]}{m + \frac{K}{A_1+(m-1)K} \left\{(m-1)+(m-2)+\ldots+(1)\right\}}$$

where E is defined as an emissivity-geometry factor for any layer and K is an average increment in surface area considered to be constant.

Conductive heat losses from the heat shield package have been calculated by finding an effecttive heat transfer coefficient for a layer of heat shielding and then considering m layers.

The expression derived for conductive loss was

$$\frac{q}{L} = \frac{2\pi (T_1 - T_m)}{\frac{m}{K_f}\left[L_m(r_{k+1}/r_k)_{avg}\right]_f + \frac{m}{K_w}\left[L_m(r_{m+1}/r_m)_{avg}\right]_w}$$

where L equals length of cylinder, r_m equals radius of m^{th} heat shield, $(r_{k+1}/r_k)_{avg}$ equals average value for the ratio of radii for adjacent heat shield layers, K equals effective thermal conductivity of a layer and the subscripts f and w refer to the foil and the spacing wire respectively.

Two computer programs have been generated to study in detail the temperature distribution throughout the complete system (nozzle losses, etc.) in greater detail. These programs are "HERS" (Heat Exchange between Radiation Shields) which was developed at GSFC, and "HIS" (Heat Transfer in an Integrated System) which was developed at G. E. "HERS" is used to study the temperature distribution through radiation shields for any given set of configuration and operating parameters. "HIS" is used to study the temperature distribution and heat loss throughout the entire integrated system.

The heat shield package for this particular thruster is comprised of a series of forty concentric Nichrome shells, 0.003 inches thick, separated by 0.014 inch diameter Nichrome wire. The shells are contained within a 0.020 inch thick cylindrical stainless steel outer shell which is welded to a flanged surface.

Fabrication of the heat shield package is accomplished through the use of a series of Aluminum mandrels varying in radius by 0.020 inches, successively. Nichrome foil 0.003 inches thick is cut to the proper dimensions, wrapped around a given mandrel and tack welded along its seam to form a cylindrical shell. Nichrome wire 0.014 inches in diameter is then wrapped around the shell and tack welded to provide for spacing between shells. The shell is slid off the mandrel until one end is flush with an end surface of the mandrel. A 0.003 inch Nichrome circular washer-shaped end piece with a pre-cut center hole is tack welded along its periphery to the shell. The Nichrome cylindrical shell, open at one end, is then removed from the mandrel and fitted over the previously made smaller shell. A second pre-cut end piece is tack welded to the open end. This procedure is continued until the required number of layers is assembled. The Nichrome shields fitted into the stainless steel housing make up the heat shield module.

A primary problem associated with the heat shield package was achieving roundness of the shells. Prior to the modular approach, shells were built up layer by layer on the thruster body without the use of Aluminum mandrels. The resulting configurations were somewhat less than round. Introduction of the mandrel technique essentially eliminated this problem and the performance with the Nichrome shells has since been repeatable.

The final selection of Nichrome as the material to use was based on performance and fabrication repeatability. The emissivity of Nichrome varies from 0.2 to 0.6 depending upon the degree of oxidation and its thermal conductivity is approximately 16 BTU/hr-ft-^oR at 2300^oR. Molybdenum has an emissivity of less than 0.1 and would be superior to the Nichrome. This was demonstrated in a laboratory model package, but fabrication problems and lack of consistent performance eliminated Molybdenum from consideration, at least for the present.

The heater module consists of a standard calrod-type swaged heater with a circular stainless steel flange welded to its base. The heater is 3.090 inches in length with a 0.375 inch diameter. The active length is 1.60 inches. The heater is constructed by wrapping a 0.010 inch diameter Platinum filament on a Magnesia core, fitting this into a Hastelloy X sheath and pneumatically impacting Boron Nitride in the free space between the core and sheath. Nominal design requirements are that 15 volts dc and 2 amps (30 watts) should produce a core temperature of 2000^{o}F. To sense this temperature a chromel-alumel thermocouple is built into the heater unit along the centerline of the Magnesia core and located at the front (nozzle) end. This heater design provides for a ΔT of less than 100^{o}F from heater centerline to the outside wall of the Hastelloy X sheath (earlier designs had temperature drops in excess of 200^{o}F). The heaters are intended to provide 10,000 hours steady state life at 2000^{o}F and 500 cycles from ambient to 2000^{o}F. The steady state life requirement has been shown practical with tests in excess of 6500 hours, but the cyclic capability is unsatisfactory. A change in the filament material introduced by General Electric shows great promise of meeting the cyclic requirement.

The thruster body module contains a sonic nozzle and provides the flow path around the heater module. The physical dimensions of this body are .475 inch diameter by 2.900 inches length. This length is not determined by the length required to heat the ammonia gas to operating temperature, but rather by the length required for the heater wire to permit operation within specified current levels. The length of the active flow path has been estimated, however, to be greater than ten (10) times that required to bring the gas up to an operating temperature of 2000^{o}F. The nozzle used for the TSK-2000-1P thruster has a throat diameter of 0.027 inches with an expansion ratio of 20 to 1. The design of this nozzle is obtained from a General Electric computer program entitled "Boundary Layer Program". This program assumes thermodynamic equilibrium flow up to the nozzle throat and frozen flow thereafter.

The thruster body and the nozzle are fabricated from Hastelloy X and the nozzle is welded to the body. Hastelloy X was selected for its temperature properties and because it acts as a catalyst promoting ammonia decomposition at high temperatures. The contour established by the computer program is not conical, it is sufficiently close that it becomes impractical to use anything but the conical shape.

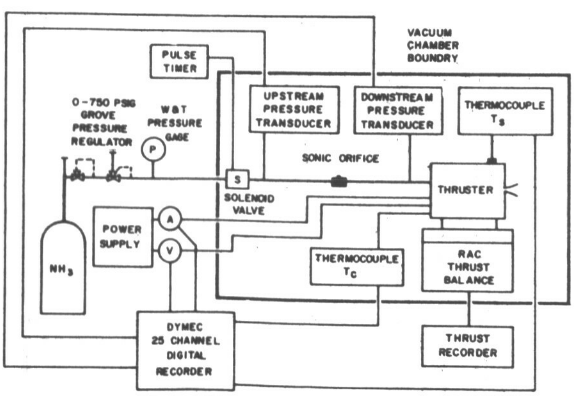
To complete the thruster body module assembly and establish proper alignment, the following procedure is used. First a heater module is inserted into the thruster shell-nozzle arrangement. Then a flange is fitted over the cylindrical body and bolted to the heater module flange. The alignment is established, and the stainless steel flange is welded to the thruster body. Perpendicularity between nozzle axis and mounting flange surface have been maintained to less than ½ degree.

When the heater module is joined together with the thruster body, the flow path is formed such that the heater sheath is the inner wall of the annulus and the thruster body the outer wall. The propellant enters the thruster assembly through the heater module flange, and flows between the heater and thruster body module flanges to the internal annulus. A conventional silicone rubber "O" ring is used to provide a seal between these two flanges. No problems have been encountered from using the "O" ring seal either by ammonia attack or from high temperature. The temperature at the seal is approximately 400°F.

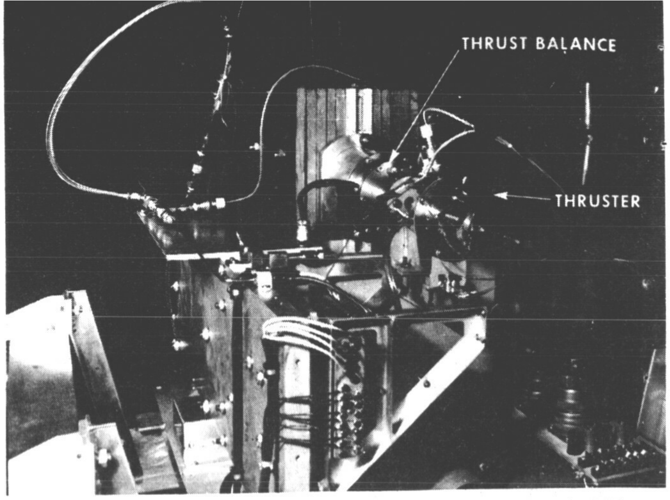
Performance

A considerable amount of test data has been generated during the development of the thermal storage resistojet both at General Electric and at Goddard Space Flight Center. The data presented here includes that obtained during the prototype qualification test program at Goddard Space Flight Center. Data from additional testing on the qualified unit is also included.

Tests are run in a 4 ft. by 5 ft. thermal-vacuum chamber. The chamber is equipped with a 32-inch diameter diffusion pump and liquid nitrogen trap. Its vacuum capability is approximately 10^{-7} Torr. Figure 3 shows schematically the instrumentation and test equipment used in the test program. Figure 4 shows the thruster mounted in the test chamber. Performance properties are determined by direct measurements of thrust, thrust chamber pressure, heater and thruster body temperatures. Propellant flow rate is calculated using the pressure drop across a calibrated orifice.



SCHEMATIC - CONSTANT MASS FLOW TESTS
FIGURE 3



RESISTOJET ON THRUST BALANCE
FIGURE 4

The thrust is measured by a Republic Aviation Corporation thrust stand system. This system uses a capacitive transudcer in an RC bridge system to measure the thrust load causing a deflection of a calibrated diaphragm spring. The range of measurable thrust is 5 x 10^{-3} pound to 4.5 x 10^{-1} pound with a basic resolution of 1 x 10^{-4} pounds. The transducers used to measure temperature, pressure, etc., are shown in Figure 3.

Figure 3 shows an orifice in the flow line between the solenoid valve and the thruster. With the orifice in the line a constant mass flow to the thruster is maintained. Tests have also been run without the orifice in an attempt to simulate constant chamber pressure performance. This type of test, however, requires that the upstream pressure regulator be manually adjusted continuously.

The prototype qualification tests were intended to serve as a basis for determining thruster reliability and acceptibility for spacecraft applications as well as to provide a measure of thruster performance characteristics. The thruster was subjected to two axes sinusoidal vibration of 10g amplitude with a frequency range of 5-2000 cps, two axes random vibration of 20 g rms with a spectral density of 0.2 g^2/cps and a frequency range of 20-2000 cps and two axes acceleration of 30 g's. No failures or significant changes in thruster performance were encountered.

Thermal vacuum tests were run with the thruster mounted in the aluminum vibration fixture by bolting a copper plate equipped with a heating and cooling coil to the bottom of the fixture. Thruster performance was evaluated with the mounting fixture at 20°C, 60°C and -10°C. Thermal vacuum test results were as expected; i.e., the thruster core temperature increased when the copper plate was heated and decreased when the copper plate was cooled.

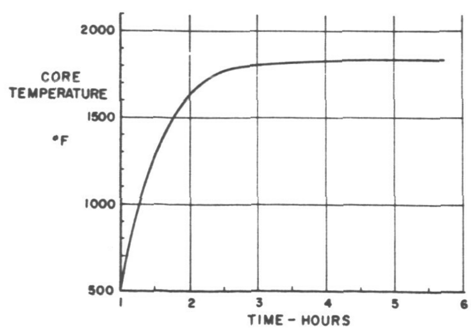
During the period in which the thermal vacuum test was run, it was found that the core temperature at 30 watts was about 50°F less than it had previously been. The mid case temperature on the outside of the thermal shield assembly was about 14°F higher. No clear conclusion has been reached as to whether some deterioration of the insulating efficiency of the thermal shielding occurred with time or possibly some damage was done to the unit during installation.

During testing, due to problems previously encountered with the thruster heaters, a number of precautions were taken to extend the life of the heater element. These were the following:

1) The power was limited to a maximum of 30 watts
2) The current was limited to a maximum of 2.2 amps
3) The power was limited to 5 watts for the first hour when heating the resistojet from ambient temperature.

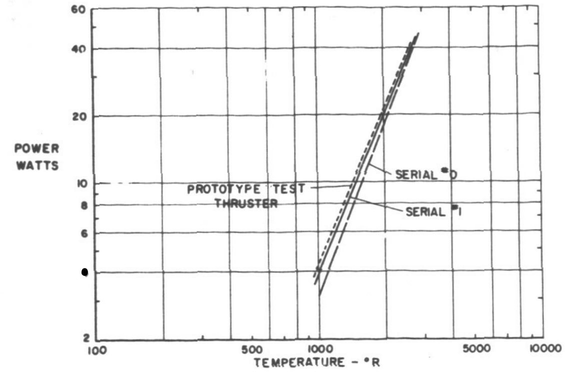
No Flow Thermal Characteristics Tests

The resistojet used in the test program was designed to provide 20 millipounds (mlb) of thrust at 22 psia chamber pressure. The initial heat up of the prototype resulted in a core temperature of 1840°F with a power input of 30 watts: 1.9 amps and 15.8 volts. As previously specified, only 5 watts were put into the thruster for the first hour. This gave a core temperature of approximately 500°F. An additional 4 hours of heating at 30 watts of power was required to reach a stabilized temperature, but 99% of the temperature change is reached in 3 additional hours. Figure 5 shows the relationship between temperature and time during heat up. If this particular unit were to be used for flight application, power usage considerations would probably dictate that the thruster be fired prior to reaching a stabilized temperature. Figure 5 shows, for example, that it takes one additional hour to heat from 500°F to 1640°F, but three more hours are required to go from 1640°F to 1840°F.



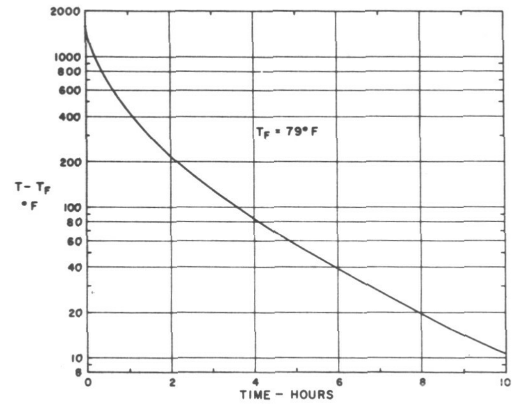
CORE TEMPERATURE vs TIME DURING HEAT-UP
FIGURE 5

A test was run to determine the stabilized resistojet core temperature as a function of power input. For this test, the power was stepped up in increments. The thruster temperature was permitted to stabilize and the amperage and voltage were recorded. The particular thruster used in these tests did not have as good a thermal storage efficiency as two previous thrusters. This is shown in Figure 6 where this thruster is compared to the two previous units (Serial Nos. 0 & 1). Testing was continued with the knowledge that this unit would give power requirements slightly higher than might be realistically achieved without any design changes.



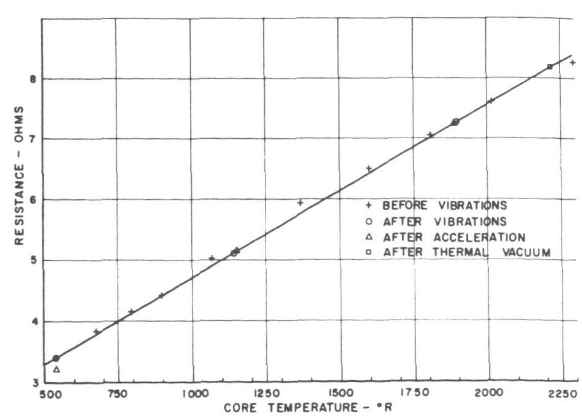
POWER vs TEMPERATURE
FIGURE 6

With the thruster core at a stabilized temperature of 1840°F, the power was shut off to determine the cool-down time of the thruster. This test gives an indication of the thermal storage capability of the thruster. Figure 7 shows the temperature drop as a function of time. Eight hours were required to reach a steady drop rate of 5°F/hour, but 90% of the temperature loss took place in the first 2 hours.



TEMPERATURE VS TIME COOLDOWN TEST
FIGURE 7

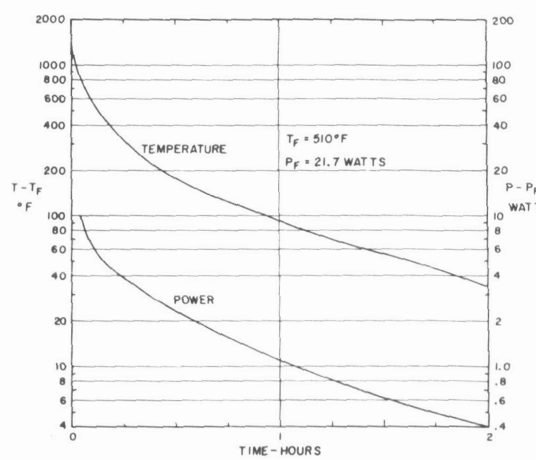
The heater electrical resistance is a prime consideration in the design of power conditioning for flight applications of the resistojet. Measurements were made both before and after vibrations, after acceleration and after thermal vacuum. Figure 8 shows heater resistance as a function of core temperature.



HEATER RESISTANCE vs CORE TEMPERATURE
FIGURE 8

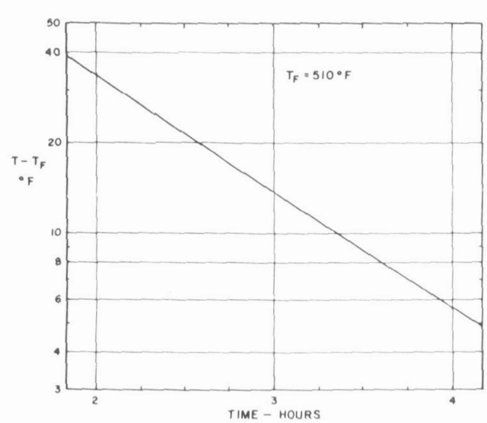
Constant Propellant Flow Tests

A continuous flow test was run for approximately four hours at a constant mass flow of 9.2 x 10^{-5} lbs/sec. The initial power, core temperature, thrust, chamber pressure and specific impulse respectively were 30 watts, 1837^{o}F, 21.1 millipounds, 22.75 psia, and 230 seconds. After the first 4 minutes of the run, the current reached 2.2 amps (due to the temperature decay) and power had to be continuously dropped to avoid exceeding 2.2 amps. At the end of 4 hours, the core temperature had reached equilibrium (dropping less than 5^{o}F/hour) at 510^{o}F with 21.7 watts input. After a little more than 3 minutes, the temperature had dropped from 1837^{o}F to 1276^{o}F. Thrust readings were inaccurate due to the large thermal drift. A thrust decay with temperature drop was expected since the mass flow was held constant. Figure 9 shows the temperature and power measurements obtained during this test.



POWER & TEMPERATURE vs TIME BASELINE FULL FLOW TES

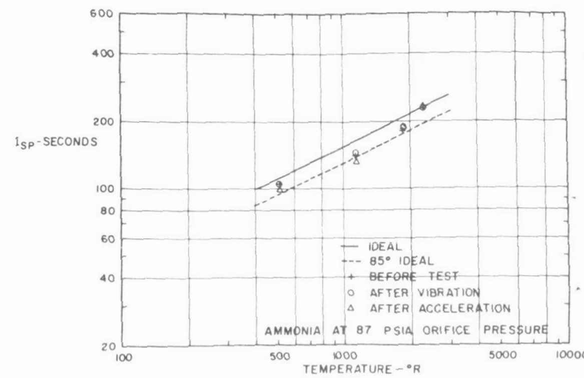
FIGURE 9 a (CONTINUED ON FIGURE 9b)



TEMPERATURE vs TIME BASELINE FULL FLOW TEST

FIGURE 9 b (CONTINUED FROM 9 a)

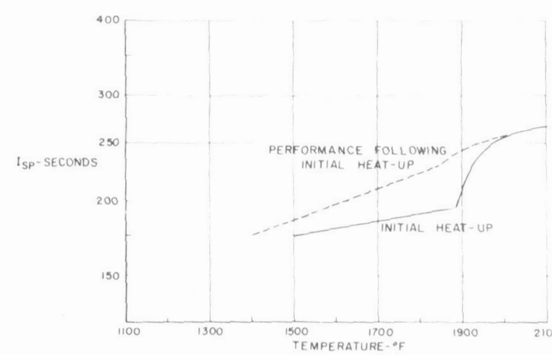
Thrust measurements were made for short pulses at various stabilized core temperatures to determine the specific impulse as a function of temperature. This relationship is shown in Figure 10.



SPECIFIC IMPULSE vs TEMPERATURE

FIGURE 10

Here it is appropriate to point out the inter relationship of I_{sp}, temperature, ammonia decompos tion and condition of the flow surface. The test thruster appeared to be new (as fabricated without oxidation) when it was received. Serial No. 1 thruster (a unit which incurred a thermal shielding failure during vibrations) had been tested before it was received and appeared to be oxidized i the nozzle cone. Figure 11 shows the I_{sp} vs. temp erature characteristics of this unit. The initial heat up had a definite break in the curve indicati that decomposition had occurred. Subsequent heatings gave higher I_{sp} values at comparable temperatures below the initial heat up decomposition temperature. This appears to indicate that the hydrogen in the dissociated ammonia gas reduced the oxic on the flow surfaces. It also indicates that some technique to enhance catalytic action is desirable. This is discussed later in the paper.



SPECIFIC IMPULSE vs TEMPERATURE SERIAL NO. I

FIGURE 11

Pulse mode tests were run to simulate possible attitude control requirements. A pulse mode of 5 seconds on and 50 seconds off was run for 5 minutes. The initial thrust was 20 millipounds at 30 watts of power with an initial chamber pressure of 22.5 psia. The ammonia flow rate was held constant at 8.9 x 10^{-5} pounds per second. Table 2 shows the performance parameters measured during the test.

TABLE 2 PULSE MODE TESTS

(50 Sec. On - 50 Sec. Off at Approximately 30 Watts Heater Power for 5 Minutes)

Pulse Number	Chamber Pressure Psia	Thrust MLB	Flow lb./sec. x 10^{-5}	I_{sp}
*1	22.0	19.5	9.1	215
2	21.5	19.2	9.1	211
3	21.25	18.9	9.1	208
4	21.0	18.6	9.1	204
5	20.75	18.1	9.1	199
**6	20.5	17.8	9.1	196

* Temperature at Beginning of Test was 1828°F.

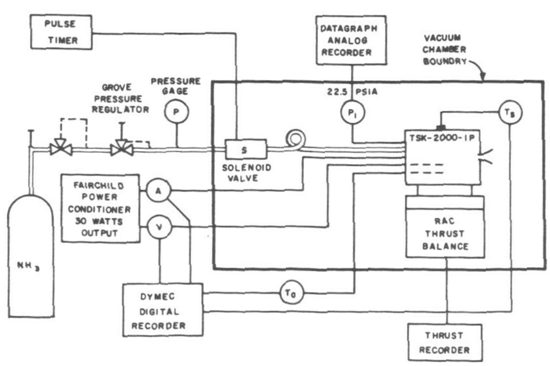
** Temperature at Completion of Test was 1725°F.

Constant Power and Constant Chamber Pressure Tests

As was expected, the constant mass flow showed a decrease in thrust as the operating temperature dropped. For spacecraft application, in order to avoid the complex attitude control logic required by a variable thrust, it is anticipated that constant chamber pressure and perhaps constant power operation will be necessary. Even with constant pressure and power, some variation in thrust would be expected with temperature decay.

A series of pulse mode tests was run to evaluate the resistojet performance at a constant thruster chamber pressure and constant power to the heater. A continuous printout of thrust and core temperature versus time was made for each test run consisting of 25-50 cycles.

Figure 12 shows the schematic setup for this particular test. The sonic orifice used in the constant mass flow tests was removed from the flow line. A power conditioner, operating on the Hall Effect principle, was incorporated to regulate the power to 30 ± 0.1 watts. This power conditioner was developed under contract for GSFC. The chamber pressure was manually regulated outside of the vacuum chamber to maintain, as near as possible, a constant chamber pressure. A pulse timer was used to actuate the flow control solenoid valve.



TEST SCHEMATIC PULSE MODE TEST
CONSTANT POWER & CHAMBER PRESSURE
FIGURE 12

During the tests, heater temperature drops as high as 930°F were experienced. This wide temperature range caused the thrust balance output and other analog readings to drift. When there was sufficient off time between pulses, the thrust balance zero was adjusted to compensate for the drift. The data collected from this test series is summarized in Table 3. The first two tests did not have sufficient time between pulses to enable the thrust balance operator to adjust the zero. The large changes in thrust measured for these two tests can be attributed at least in part to thermal drift since later tests, in which there was a larger drop in temperature, showed less change in thrust. A further indication that thermal drift of the thrust balance zero causes an error in readings was found in the 30 seconds on - 170 seconds off test. Here the first seven pulses gave thrust readings of 22.3 mlb or better. Afterwards except for one reading of 22.4 mlb, the readings ranged from 21.6 to 22.1. The early pulses are the ones in which the largest changes are occurring in the temperature of the thruster.

Conclusions and Expected Design Improvements

The development of the thermal storage resistojet has progressed to the point where it can be seriously considered for application to spacecraft attitude control, station keeping and station seeking. A prototype unit has successfully completed vibration and thermal vacuum testing similar to that which might be required of a flight unit. The resistojet performance, although presently below design goals, is a considerable improvement over cold gas systems.

Design and performance improvements that are expected and have been demonstrated to a degree in the laboratory include:

1) Weight reduction
2) Lower power input requirement for equivalent operating temperature
3) Improved I_{sp} at lower temperatures through more complete and reliable ammonia decomposition.

With respect to the thruster weight, a significant improvement can be achieved through redesign of the mounting flange. In the model TSK-2000-1P thruster, approximately one-half the total weight is in the mounting flange. Changing the flange material to Titanium, for example, would reduce the thruster weight about 25%. The use of an isolated heater element would eliminate one of the flange sections.

POWER AND THRUSTER CHAMBER PRESSURE PULSE TESTS

Number of Pulses	Seconds on	Seconds off	Duty Cycle %	Thrust Chamber Pressure Psia	Minimum Recorded Thrust mlb	Maximum Recorded Thrust mlb	Core Temp. at Start of Test °F	Core Temp. at End of Test °F	Heater Power Watts
50	5	5	50	23	20	21.7*	1761	1153	30±0.1
50	2	11 1/3	15	22.5	20.5	21.8*	1774	1453	30±0.1
50	30	170	15	22	21.6	22.7	1774	1369	30±0.1
50	15	15	50	23.0**	20.0	21.0	1777	930	30±0.1
25	30	30	50	23.0	20.8	21.7	1769	947	30±0.1
25	60	60	50	23.5	21.1***	21.7	1767	837	30±0.1

* Duty cycle did not permit adjustment of thrust balance zero setting between pulses.

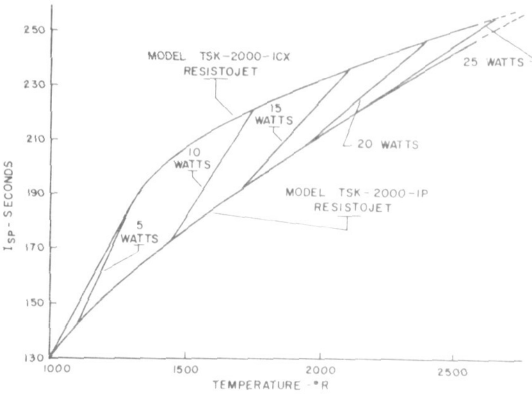
** Chamber pressure dropped during test reaching a minimum of 22.2 after 9.5 minutes.

*** Except for the first two readings, all other readings were between 21.5 and 21.7.

The power requirement to achieve an operating temperature of 2000°F could be improved on the order of ten (10) watts by thermal isolation of the heater element according to an analysis performed at NASA.

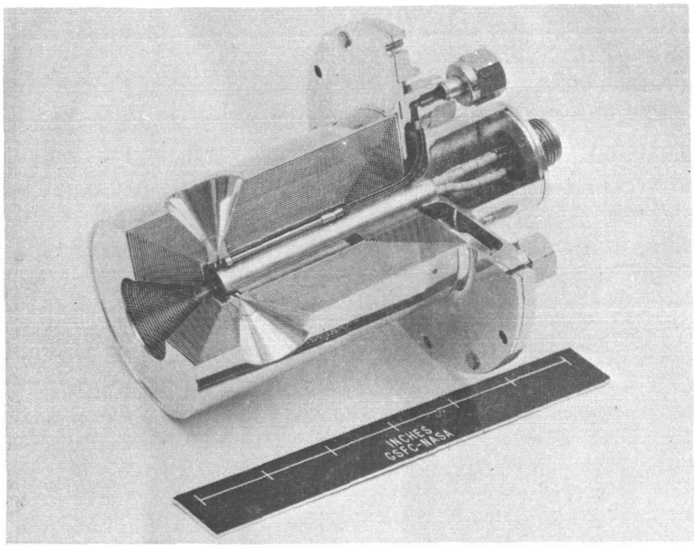
To improve the specific impulse property of the thruster a sub-contract was established with Englehard Industries through General Electric to study and test catalytic compounds. As a result of this contract a catalytic compound (proprietary to Englehard) was tested that assures a minimum of 90% ammonia decomposition at 1500°F with no appreciable degradation as a function of time. This catalyst is in the form of 20-25 mil alumina spheres with the active catalyst coating the spheres. The thruster flow annulus would be filled with these spheres such that they would be in intimate contact with the gas flow.

A model TSK-2000-1P thruster was modified to incorporate the catalytic compound and provide a thermally isolated heater. This thruster was designated as a model TSK-2000-1CX thruster. Laboratory test results for this thruster were compared with the TSK-2000-1P thruster. Results of this comparison are shown in Figure 13 provided by General Electric. As an example of the comparison for the two thrusters, consider a fifteen watt input. For the model TSK-2000-1P thruster this corresponds to a core temperature of 1700°R and an I_{sp} of 192 seconds. For the TSK-2000-1CX thruster the core temperature would be 2100°R with an I_{sp} of 236 seconds.



POWER AT CONSTANT TEMPERATURE OR TEMPERATURE AT CONSTANT POWER (COURTESY G.E.)
FIGURE 13

Long term earth orbital and planetary missions are among the many potential applications for the resistojet. One particularly attractive application of the resistojet would be the use of such a system in the form of multijet thrusters (each consisting of four or five jets having a common heater) for orbit adjustment and attitude stabilization of earth synchronous spacecraft. Such a thruster has been designed, fabricated and tested. A sectioned view of the unit is shown in Figure 14. The design permits sizing of attitude control jets and orbital adjustment jets to individually required thrust levels. Preliminary studies for an applications technology satellite compared the total system weight (propellant tangage and feed subsystem, propellant, thrusters, etc.) of this type of multijet system to that of a cold gas Freon system. The ammonia resistance jets would decrease the total system weight requirement from that of a Freon system of comparable total impulse (17000 lb-sec.) by a factor of four (4) (650 lbs. to 165 lbs.) even if an average I_{sp} of only 150 seconds could be attained with the resistojets.



SECTION - MULTIJET THRUSTER
FIGURE 14

In addition to the design improvements previously mentioned, the next major step in the resistojet development is to integrate the thruster with a propellant storage and feed subsystem capable of meeting the specific mission requirements of future spacecraft missions.

Acknowledgment

The authors would like to acknowledge that the resistojet testing described in this paper was performed under contract by Melpar, Inc. at GSFC. Design information and performance data has been borrowed freely from the Melpar and G. E. reports. It is impossible to give particular credit to the many people at G. E., Melpar, and GSFC who have contributed to the design, development, and testing of the thermal storage resistojet.

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