1231

Performance and Endurance Tests of a Multipropellant Resistojet for Space Station Auxiliary Propulsion

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Abstract

This paper presents the results of an effort by the NASA Lewis Research Center (LeRC) to demonstrate the technology readiness of a long-life multipropellant resistojet for Space Station auxiliary propulsion. Experiments were performed to evaluate the compatibility of grain-stabilized platinum tubes at temperatures up to 1400 °C in environments of CO_2, CH_4, NH_3, H_2O, and H_2. All samples tested showed extrapolated lifetimes in excess of 10,000 hr based on 10 percent mass loss as end-of-life. However, samples tested in an ammonia atmosphere at 1400 °C showed severe pitting, which raised concerns about the compatibility of grain-stabilized platinum with ammonia-containing atmospheres. Additional tests showed that reducing the metal temperature to about 900 °C (±100 °C) significantly reduced this adverse effect. Characterizations of a simple, radiatively-coupled design fabricated from grain-stabilized platinum on a variety of fluids were performed. Propellant gases used included hydrogen, helium, methane, air, nitrogen, argon, and carbon dioxide. Thrust levels observed ranged from 90 mN to 420 mN at input electric power levels from 140 to 240 W. Measured specific impulse values ranged from 100 to 400 sec. Endurance testing of the same type of thruster using CO_2 propellant was begun, and at the time of this writing 1400 operating cycles with on-time of about 1160 hr have been achieved.

Introduction

During the 1960's there was an active technology effort to develop resistojets for the Manned Orbital Research Laboratory (MORL) using Environmental Control and Life Support Systems (ECLSS) effluents. The MORL program, and the associated biowaste resistojet technology work,[1-4] was terminated in the early 1970's. Over the past six years, high-performance hydrazine resistojets have performed North-South stationkeeping for Intelsat and RCA spacecraft in geosynchronous orbit, marking a renewed interest in resistojet technology. Two thruster developers have provided approximately one hundred resistojet thrusters for twenty-six flights.[5,6] Resistojets have also been considered for auxiliary propulsion of spacecraft operating at low earth orbit. Recently, a resistojet technology program was reinstated by NASA to focus on materials evaluation, fabrication methods, performance, and life assessments for Space Station application. The technology development goals are to emphasize thruster life and multipropellant capability rather than performance.

The Lewis Research Center (LeRC) is actively involved in propulsion component technology in support of Space Station Advanced Development Program.[7] The resistojet is one of the propulsion options being investigated[8] due to its unique ability to provide thrust for atmospheric drag make-up while disposing of a wide variety of excess fluids expected to be present on board Space Station. The resistojet can vent these excess gases propulsively or nonpropulsively, insuring no change of phase occurs during venting. Significant Space Transportation System (STS) costs would be associated with the disposal of unwanted fluids from Space Station. These costs would be reduced substantially with the use of resistojets for waste fluid disposal. Recent studies have explored these and other potential benefits of propulsion systems which use low-thrust resistojets.[9-10]

In an effort to reduce Space Station servicing costs associated with thruster replacement, emphasis has been placed on thrusters designed for long life rather than high performance. To this end, the selected thruster designs utilize a structurally rugged heater/heat exchanger assembly. The design life goal is 10 000 hr for thrusters operating on hydrogen, helium, methane, nitrogen, air, argon, and carbon dioxide at specific impulse values ranging from 100 to 500 sec. Sources and types of Space Station fluids suitable for use as resistojet propellants are itemized in Table I.

The objective of this program was to develop a laboratory model resistojet which would facilitate verification of performance predictions, thruster material/propellant compatibility, weld integrity, and provide preliminary lifetime data for operation in a thermally cyclic mode. A simple, rugged laboratory model resistojet was fabricated from grain-stabilized platinum. This thruster design was characterized on a variety of propellant gases at heater temperatures of 1400 °C. The same type of thruster has begun endurance testing using CO_2 propellant. A more refined thruster with improved long life characteristics has been designed under contract with LeRC by the Rocketdyne Division of Rockwell International and Technion.[11] These thrusters will be tested for performance and endurance at LeRC, and will also be sent to Marshall Space Flight Center for test-bed evaluation.

This paper will discuss briefly the rationale for the material chosen for the multipropellant resistojet. The results of performance evaluations of the laboratory model resistojet operating on hydrogen, helium, methane, air, nitrogen, argon, and carbon dioxide will then be presented. Finally, the progress of an ongoing endurance test of the laboratory model resistojet using CO_2 propellant will be reviewed.

Apparatus and Procedure

Thruster Design Description

A radiatively-coupled heat exchanger was chosen for the laboratory model thruster to separate the heating element from the propellant, thus minimizing the effects of any potential propellant/heater material interaction. The heating element is made from a 2.03 mm o.d. by 1.52 mm i.d. coiled tube and

is placed inside a concentric tube heat exchanger (Fig. 1). The tubular heater is comprised of 22 turns over a length of 5.82 cm. To reduce heat losses from the rear of the thruster, the propellant flow path was lengthened near the propellant inlet by channeling the flow circumferentially around the heat exchanger body. The flow is then directed axially toward the nozzle after passing through 16 small channels contained within the forward (hottest) portion of the heat exchanger. The hot gases then exit through the nozzle.

The multipropellant thruster assemblies tested were fabricated entirely from zirconia grain-stabilized platinum. The coiled tube heaters were also made from grain-stabilized platinum. Components were joined using electron beam welding. Design characteristics of the thrusters are summarized in Table II.

The thrusters were wrapped with radiation shielding consisting of two layers of 0.025 mm platinum foil followed by 13 layers of 0.127 mm stainless steel foil. The layers of shielding were separated by small diameter wires.

Measurement Apparatus

The resistojets were characterized using a thrust stand derived from a TRW-designed device which uses the horizontal displacement of a mounting plate as an indication of applied thrust.[6] Figure 2 illustrates this thrust stand schematically. The mounting plate is made from a machinable ceramic material to reduce heat transfer from the thruster mounts to the rest of the thrust stand, and is supported on four flexures made from stainless steel shim stock. Displacement of the mounting plate is monitored using a linear variable differential transformer (LVDT). The output of the LVDT was calibrated against thrust using small weights (approximately 148 mN each) hung from a length of cord. A low-friction pulley transfers the vertical force exerted on the cord by gravity into a horizontal (thrust) force. The magnitude of the calibration thrust was changed by varying the number of calibration weights supported by the cord. This was accomplished using a movable weight pan.

Electrical power was supplied to the thruster by two 0.32 cm diameter copper rods. Propellant was fed to the thruster through a length of 0.32 cm o.d. thin-walled stainless steel tubing. A second tube (identical to the propellant feed line) connected the thruster inlet to a pressure gauge. The propellant feed tube and the pressure tap tube were shaped in a sine-wave configuration to minimize their lateral stiffnesses, thus increasing the sensitivity of the thrust stand.

The thruster mounting plate and LVDT were submerged in an oil bath to minimize thermal expansion of the plate and the associated thermal drift in the LVDT output. This thrust system yielded a standard deviation in indicated thrust of approximately 1 mN during calibration. Thermal drift during operation at the power levels required was minimal. This drift was negated by allowing the thruster to establish equilibrium at each operating point, then turning off the propellant flow to cancel thrust, allowing a new thrust stand zero reading to be established. The thruster was assumed to be at equilibrium when the inlet pressure, heater voltage, and heater current readings

stabilized. Figure 3 shows a photograph of the multipropellant resistojet mounted on the thrust stand, ready for performance testing.

Power Processing

Heater power was supplied by a dc power supply with an output capacity of 50 V at 50 A. This unit is operable in both voltage-and current-limiting modes. The dc power supply was used in conjunction with a dc resistance controller designed at LeRC which allowed constant heater resistance operation of the thruster. Heater power consumption was calculated from measurements of current and voltage at the thruster heater terminals.

Propellant Flow Measurement

Propellant flow rates were measured using transducers which are thermally sensitive to gas mass flow rate and heat capacity. These flow transducers were calibrated for output reading versus actual mass flow rate for each of the propellants used in the multipropellant resistojet. The calibration curves obtained were linear over the flow rate ranges tested. The uncertainty in the measured values of the propellant gases is estimated to be better than 3 percent for all but the lowest flow rates measured.

Test Chamber

The characterizations of the multipropellant resistojets were conducted in a vacuum chamber measuring 4.6 m in diameter and 19 m long,[12] and utilizing a pumping train consisting of twenty 0.8 m diameter oil diffusion pumps backed by four positive displacement, lobe-type rotary blowers and four oil-sealed rotary piston vacuum pumps. With no flow into the tank this pumping train can achieve pressures of approximately 4×10^{-7} torr. Tank pressures of less than 6×10^{-4} torr are attainable even at hydrogen flows of 10 standard liters per minute. Data relating some flow rates and the corresponding tank pressures for hydrogen and nitrogen are summarized in Table III.

Material Compatability Testing

The requirement that the Space Station resistojet be capable of operation for extended periods of time on a variety of propellants is extremely stringent, since the range of fluids of interest includes both oxidizing and reducing gases. The material chosen for this application was grain-stabilized platinum. This platinum-based material was believed to have an adequate combination of high-temperature strength and corrosion resistance in both oxidizing and reducing atmospheres. The grain stabilization is desired to minimize grain growth which occurs when materials are held at high temperatures for extended periods of time. Unfortunately, published information on the long-term compatibility of grain-stabilized platinum with the propellants of interest at 1400 °C was not available. Therefore, a study was conducted to evaluate the effects of long-term exposure of high-temperature grain-stabilized platinum tubes to environments of carbon dioxide, methane, hydrogen, ammonia, and steam.[13-14]

Two facilities were used in the grain-stabilized platinum compatibility tests. Figure 4 shows the test chamber used in the tests with the flowing H_2, CH_4, and the CO_2/CH_4 mixtures. A schematic of the apparatus is shown in Figure 5. The 2.54 cm o.d. by 20.3 cm chamber was constructed of a quartz tube with metalized ends. The test chamber was mounted in the bell jar of a vacuum tank. A mechanical roughing pump was used in purging the chamber and maintaining the cell pressure. A current-controlled dc power supply with a 100 A, 100 V maximum capacity was used in each experiment. The test piece was coiled from 0.203 cm o.d. by 0.025 cm wall tubing. It was supported by 0.635 cm diameter power leads. The platinum-yttria experiment in ammonia incorporated a similar test cell, except the chamber was stainless steel with a quartz viewing window.

An additional experimental facility employed a 45.7 cm diameter stainless steel bell jar. This bell jar was used for platinum-yttria coil tested in stagnant CO_2. A roughing pump was used in initial purging and in test shutdown procedures. The same type of power supply and similar specimen mounting arrangement were used in each experiment.

The grain-stabilized platinum tubes were first vacuum annealed at 1000 °C for 30 min. This annealing made the tube specimens ductile enough to be coiled. Each coiled tube was cleaned with NA-500 (a degreaser), then rinsed with ethyl alcohol and dried with high-purity nitrogen gas. The mass of each coil was then measured using an analytical balance readable to 0.1 mg.

A series of experiments to determine the compatibility of a grain-stabilized platinum with H_2, CH_4, NH_3, CO_2, CH_4, and H_2O was performed. Each coiled tube was maintained at high temperature (500 to 1400 °C) in a flowing gas environment at ambient pressures of about 1.4 atm. Gas flow rates through the test cells were approximately 100 standard cm^3/min. Samples tested in H_2, CO_2, and H_2O were heated to 1400 °C, while samples tested in CH_4-containing environments were heated to about 500 °C to avoid CH_4 dissociation. Another test was performed in a stagnent CO_2 environment at a coil temperature of about 1300 °C. Test cell pressure for this test was about 0.7 atm.

Performance Test Procedure

Performance of the thrusters was measured for each of the seven propellants over a range of thrust levels at constant heater temperature. The constant heater temperature performance was obtained using a dc resistance controller. Heater temperature for all tests was 1400 °C, and was determined based on a temperature-resistance calibration obtained using a two-color optical pyrometer.

Endurance Testing

One of the two laboratory model resistojets made from grain-stabilized platinum was installed in a small vacuum chamber (Fig. 6) for endurance testing using CO_2 propellant. The propellant mass flow was held constant at approximately 0.125 g/s, while the heater power was cycled at a frequency of one cycle per hour with a duty cycle of 83 percent.

The heater temperature at steady state was approximately 1400 °C. This type of operation causes periodic fluctuations in both thermal and mechanical stresses.

Operating parameters monitored included heater voltage and current, propellant mass flow rate, feed system and thruster inlet pressures, test chamber pressure, and outside radiation shield layer temperature. Thrust was not monitored in an effort to reduce test apparatus complexity.

RESULTS AND DISCUSSION

Material Compatability Results

The results of the material compatibility experiments, summarized in Table IV, indicate that grain-stabilized platinum tubes are compatible with carbon dioxide, methane, hydrogen, and steam under the test conditions. The extrapolated lifetimes, based on ten percent mass loss as end of life, are well in excess of the design life goal of 10 000 hr in all cases. Figure 7 shows two cross sections of one of the grain-stabilized platinum test specimens: the first was taken before testing; the second was taken after 2000 hr of exposure to CO_2 at a coil temperature of 1300 °C. It is apparent that no significant grain growth has occurred, and that attack of the sample surfaces by the environment are minimal. These photos are typical of the results observed for specimens tested in CO_2, H_2, CH_4, and steam.[12-13] The sample tested in ammonia, however, showed pitting that extended well into the material, which could affect the material strength and indicate a potential interaction between grain-stabilized platinum and hydrazine decomposition gas products. Reducing the operating temperature of the platinum surfaces to about 900 °C when exposed to ammonia-containing atmospheres resulted in a significant reduction in such pitting.

Based on this investigation, grain-stabilized platinum should be suitable for use over extended periods of time at high temperatures for the fluids tested, although it may be necessary to limit the material temperature in ammonia environments to less than 1000 °C.

Characterization Test Results

Thruster evaluation was undertaken at a power level of about 200 W for thrust levels ranging from 90 to 420 mN. Specific impulse, thrust, and input power parameters are reported for each of the seven propellants. The data were also reduced to examine the values of overall efficiencies.

With this simple, radiatively-coupled thruster design, large temperature drops between the thruster heater and heat exchanger were expected. Estimates of the temperature difference between the heater and heat exchanger range from 360 °C for low-thrust operation on argon propellant to 1290 °C for operation on hydrogen at high thrust levels. These estimates were based on the assumption that the wall temperature was approximately equal to the estimated final gas temperature. The gas temperatures were calculated by performing an energy balance on the gas as it passed through the heat exchanger. The power in the gas leaving the heat exchanger,

P_{gf}, is the thrust power divided by the square of the nozzle efficiency:

$$P_{gf} = \frac{T*I_{sp}*g}{2\eta_{nozzle}^2} \qquad (1)$$

Where T and I_{sp} are the measured values of thrust and specific impulse, respectively, and η_{nozzle} is the nozzle efficiency, defined as the ratio of the actual specific impulse to that predicted by one-dimensional isentropic flow equations for a nozzle with the same area ratio as the one being investigated.

The power in the gas entering the heat exchanger, P_{go}, is:

$$P_{go} = \dot{m}h_o \qquad (2)$$

Where $\dot{m}$ is the mass flow rate and h_o is the cold (300 K) gas enthalpy. The power added to the gas within the heat exchanger is the difference between P_{gf} and P_{go}, and can be equated to the product of the mass flow rate and the enthalpy rise of the gas:

$$\dot{m}(h_f - h_o) = \frac{T*I_{sp}*g_e}{2\eta_{nozzle}^2} - \dot{m}h_o \qquad (3)$$

from which the enthalpy of the gas leaving the heat exchanger, h_f, can be calculated. This value was then used with standard gas property tables[15] to determine the final gas temperature. The value of the nozzle efficiency was assumed to be 0.95 based on estimates used previously by Zafran and Jackson.[16]

Figure 8 shows the ranges of thrust levels and specific impulse values observed for the multipropellant resistojet for the various propellants at heater temperatures of approximately 1400 °C. This heater temperature was chosen due to heater material temperature limitations. Because of the configuration of the heater element, this temperature limitation caused maximum power dissipation from the heater to be in the vicinity of 200 W. Flow rates for all propellants ranged from 0.08 to 1.6 kg/hr at stagnation temperatures from approximately 110 °C for hydrogen at high flow rates to about 1040 °C for argon at low flow rates.

Figures 9 through 11 illustrate the variations of specific impulse and input electric power with thrust for the seven propellants tested at heater temperatures of approximately 1400 °C. Thrust levels were varied by changing the propellant mass flow rate. Power consumption varied approximately as the square root of thrust.

The maximum specific impulse values obtained ranged from a high of 402 sec for hydrogen to a low of 133 sec for argon. The high values of specific impulse measured for the remaining five gases were 278, 180, 154, 148, and 122 sec for helium, methane, nitrogen, air, and carbon dioxide, respectively. Specific impulse values at 300 K are also shown in Figures 9 through 11.

Figure 12 shows the ranges of overall efficiencies calculated for the multipropellant

resistojet for heater temperatures of 1400 °C for the seven propellant fluids. The specific impulse values at which the maximum and minimum efficiencies were calculated are also given. The maximum efficiencies generally correspond to the minimum specific impulse values. Similar resistojet characterizations performed at higher heater temperatures using hydrogen, ammonia, and nitrogen propellants were recently reported by TRW[16] and Rocket Research.[17]

Endurance Test Progress

The laboratory model resistojet made from grain-stabilized platinum is currently undergoing endurance testing on carbon dioxide propellant. Thruster inlet pressure varies from 0.12 to 0.19 MPa, corresponding to thrust levels on the order of 93 to 147 mN. The maximum temperature on the outer layer of the radiation shield reaches about 300 °C at equilibrium under power, and drops to about 170 °C during the cool down period of each cycle.

Difficulties encountered thus far have been confined to the heater element. The heater design, originally conceived for use in a rhenium thruster, does not provide sufficient rigidity at high temperature to withstand gravitational stresses. This resulted in a tendency for the heater to sag, contacting the inside of the thruster body and causing a short circuit. This problem was remedied by installing a heater support structure (see Fig. 1) made of ion sputtered or vapor-deposited platinum on alumina. While interactions between platinum and alumina at high temperatures have been observed in the past[18] to cause degradation of the platinum structure, preliminary tests suggest that applying a thin coating (nominally 2000 Å) of platinum retards this interaction. This support structure allowed about 15-1/2 of the 22 turns of the heater coil to view the inside wall of the heat exchanger.

As of this writing this thruster has completed more than one-half of the planned 2000-hr (2400 thermal-cycles) endurance test using carbon dioxide propellant. As shown in Figure 13, the heater voltage has dropped from an initial value of 6.7 V to approximately 5 V with a variation of about 0.3 V on an intermittant basis. The reason for the variation is believed to be the occasional opening and closing of a contact point between two or more of the turns of the heater coil. Heater current is held constant at 29.0 A during the 83 percent of each 1 hour cycle that power is supplied to the heater. Thus the reduction in heater voltage has caused a drop on the electric power consumed by the heater from 194 W to approximately 145 W. However, Figure 14 shows that the change in the estimated specific impulse over the course of the life test has varied by less than 10 percent. These estimates were made by comparing the ratio of the electric power to the mass flow rate for the life test thruster to a curve calibrating specific impulse to this ratio. The data used to generate this curve was obtained prior to the beginning of the life test.

The goals of this effort were to evaluate the compatibility of the grain-stabilized platinum thruster material with one of the fluids of interest for an extended period of time under conditions which one would expect to be encountered during normal use.

In addition, information about the effects of cyclic thermal and mechanical stresses on the material itself as well as the welded joints could be gained.

In future efforts, an engineering model multi-propellant resistojet supplied by the Rocketdyne Division of Rockwell International and Technion will be evaluated for performance and endurance in a manner similar to that described in this paper. This thruster design will eliminate the possibility of heater sag by using a sheathed heater element supported by a rugged heat exchanger body.[11]

Concluding Remarks

Resistojet thrusters capable of operation for extended periods of time on a wide variety of fluids are candidates for Space Station auxiliary propulsion due to their unique ability to provide atmospheric drag make-up while disposing of excess fluids expected to be present onboard Space Station. Resistojets can vent unwanted fluids propulsively or non-propulsively, thereby reducing transportation costs associated with removing waste fluids from Space Station orbit.

Experiments were performed to evaluate the compatibility of grain-stabilized platinum tubes at temperature, up to 1400 °C in environments of CO_2, CH_4, NH_3, H_2O, and H_2. All samples tested showed extrapolated lifetimes in excess of 10 000 hr based on 10 percent mass loss as end-of-life. However, samples tested in an ammonia atmosphere at 1400 °C showed severe pitting, which raised concerns about the compatibility of grain stabilized platinum with ammonia-containing atmospheres. Additional tests showed that reducing the metal temperature to about 900 °C(±100 °C) significantly reduced this adverse effect.

In an effort to demonstrate the technology readiness of the long-life multipropellant resistojet, laboratory model resistojets fabricated from grain-stabilized platinum were subjected to endurance testing using carbon dioxide and were characterized on hydrogen, helium, methane, air, nitrogen, argon, and carbon dioxide. Thrust levels observed during characterization ranged from 90 mN to 420 mN at input power levels ranging from 140 W to 240 W. Specific impulse values calculated ranged from 96 to 402 sec. The endurance test has completed more than 1400 one hour cycles with a heater duty cycle of 83 percent.

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TABLE I. – POTENIAL WASTE FLUID SOURCES FOR SPACE
STATION AUXILIARY PROPULSION

Space Station Subsystem	Waste Gases/Fluids
ECLSS	
Bosch CO_2 reduction	H_2, H_2O
Sabatier CO_2 reduction	CO_2, CH_4
Lab Modules	
Material, life science	Ar, CO_2, He, H_2
	N_2, H_2O
Attached Payloads	To be determined
Miscellaneous purge gases	N_2, Ar, H_2
Orbit transfer vehicle tank farm	GO_2, GH_2

TABLE II. – MULTIPROPELLANT RESISTOJET DESIGN CHARACTERISTICS

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Shell/Nozzle:
    Material . . . . . . . . . . .    Grain-stabilized platinum
    Nozzle throat diameter, mm . . . . . . . . . . . . . 0.84
    Nozzle area ratio  . . . . . . . . . . . . . . . . . . 82
    Nozzle half angle, deg . . . . . . . . . . . . . . . . 20
    Body length, cm  . . . . . . . . . . . . . . . . . . 13.0
    Body diameter, cm  . . . . . . . . . . . . . . . . . 1.92
Heat Exchanger:
    Material . . . . . . . . . .    Grain-stabilized platinum
    Number of channels . . . . . . . . . . . . . . . . . . 16
Heater Element:
    Material . . . . . . . . . .    Grain-stabilized platinum
    Coil length, cm  . . . . . . . . . . . . . . . . . . 5.82
    Coil pitch . . . . . . . . . . . . . . . . . . . . . 0.10
    Coil diameter, cm  . . . . . . . . . . . . . . . . . 1.04
    Maximum operating  . . . . . . . . . . . . . . . . . 1400
      temperature, °C
    Design life, hr  . . . . . . . . . . . . . . . . 10 000
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TABLE III. – VACUUM TANK 5 PUMPING CAPABILITY USING HYDROGEN AND
NITROGEN WITH TWO PUMPING CONFIGURATIONS

Pumping configuration	Hydrogen		Nitrogen	
	Flow rate g/s	Pressure[a] torr	Flow rate g/s	Pressure[a] torr
20 Oil diffusion pumps	0.02	7×10^{-4}	0.1	5×10^{-4}
4 Lobe blowers and 4 oil sealed rotary	0.02	6×10^{-2}	0.1	4×10^{-2}

[a]Pressure measured with cold cathode ionization gage with readings corrected for gas type.

TABLE IV. – SUMMARY OF GRAIN STABILIZED
PLATINUM EXPERIMENTS

Propellant	Coiled heater temp., °C	Heater initial mass, g	Coiled heater mass loss, g[a]	Extrapolated life,[b] hr
Platinum – Yttria				
CO_2	1400	9.0194	0.0030	300 000
CH_4	500	12.6384	.0008	1 500 000
H_2	1400	12.6589	.0062	200 000
NH_3	1400	12.5982	.0055	200 000
H_2O	1400	13.0695	.0116	113 000
Platinum – Zirconia				
CO_2	1400	13.1955	0.0016	800 000
CH_4	500	11.6969	.0000[c]	1 000 000
H_2	1400	13.2093	.0031	400 000
NH_3	1400	13.0632	.0066	200 000
H_2O	1400	11.5133	.0245	45 000

[a]After 1000 hr operation.
[b]Time to 10 percent mass loss.
[c]0.0001 g, accuracy of balance.

PROPELLANT
FLOW →

ALUMINA

GRAIN-STABILIZED PLATINUM

20°

PROPELLANT INLET

FLOW SPIRALING CHANNEL

PLATINUM-COATED ALUMINA HEATER
COIL SUPPORT STRUCTURE

COILED HEATER

POWER LEADS

INNER TUBE (HEAT EXCHANGER) AND HEATER

HEAT EXCHANGER CHANNELS (16)
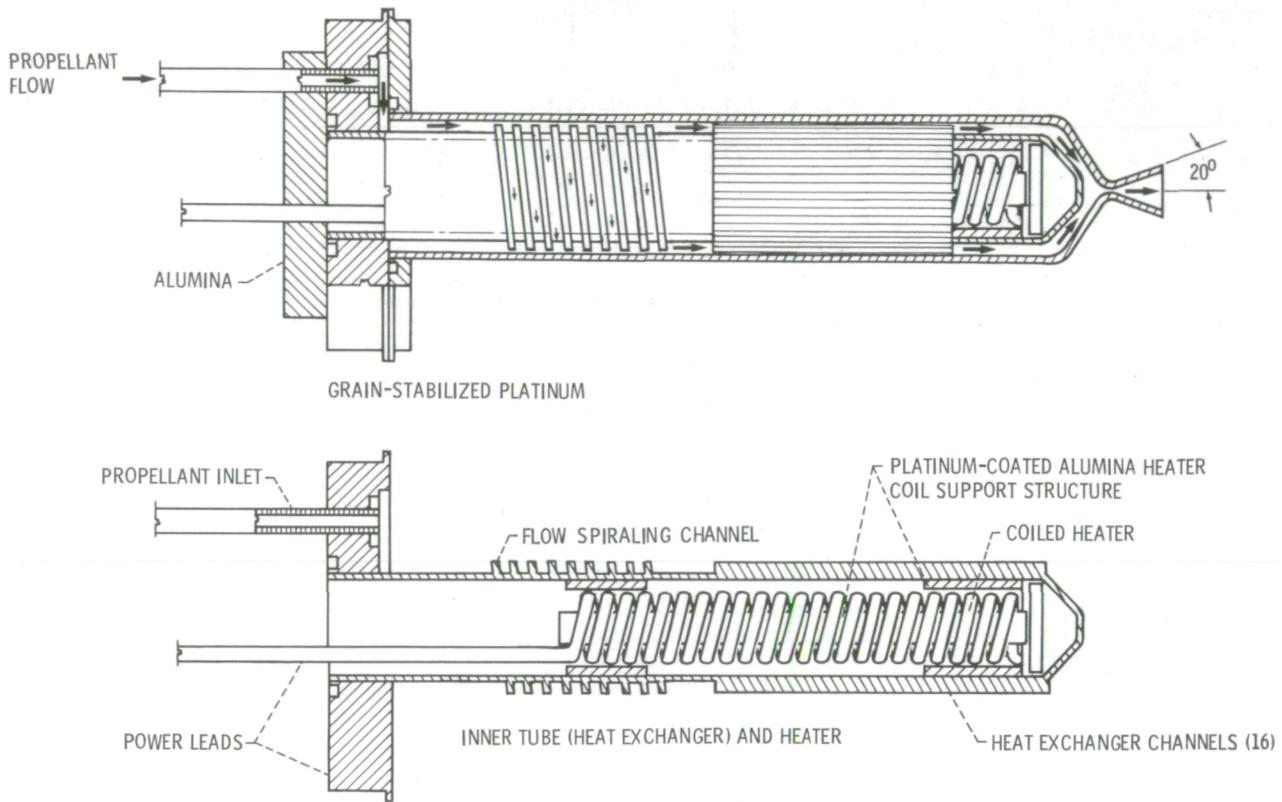
Figure 1. - Multipropellant resistojet cross-section.

TO PRESSURE GAUGE

PROPELLANT IN

POWER LEADS

PROPELLANT AND PRESSURE TAP LINES (FLEXURES)

RESISTOJET

LOW FRICTION PULLEY

RADIATION SHIELD

OIL LEVEL

CALIBRATION WEIGHT

FLEXURES

LVDT

BASE PLATE

THRUSTER MOUNTING PLATE

MOVABLE WEIGHT PAN

FLEXURES
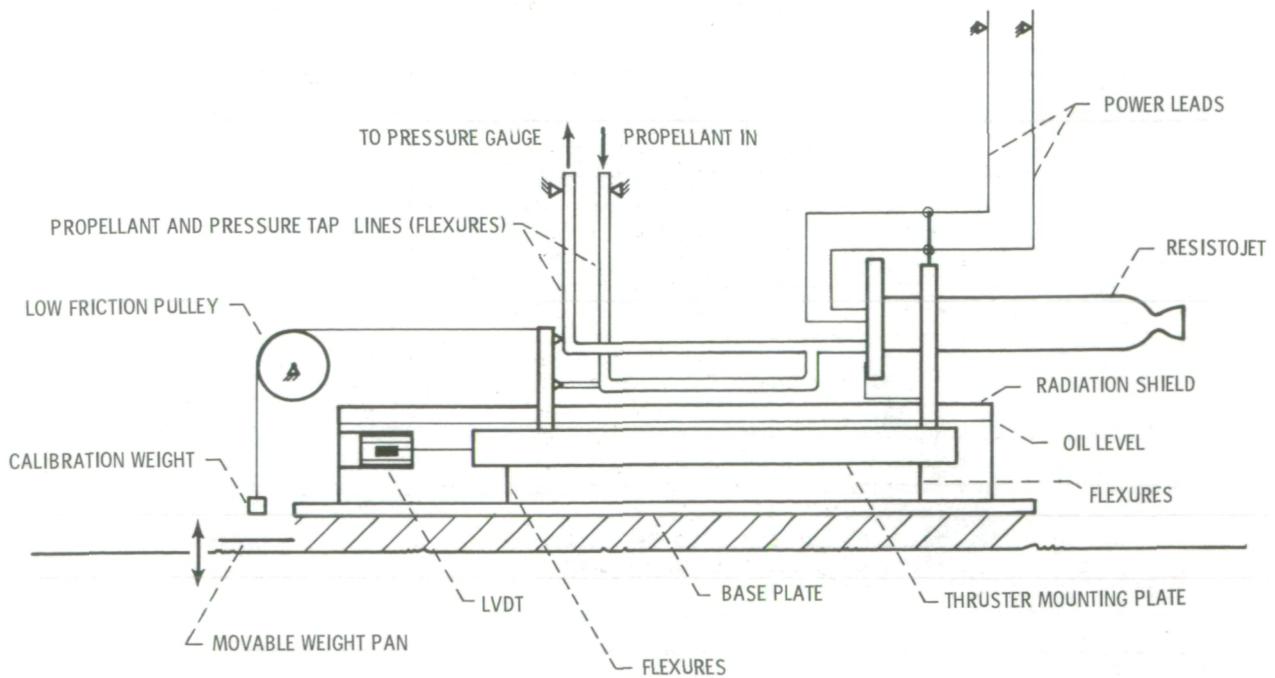
Figure 2. - Thrust stand schematic.

C-85-6460
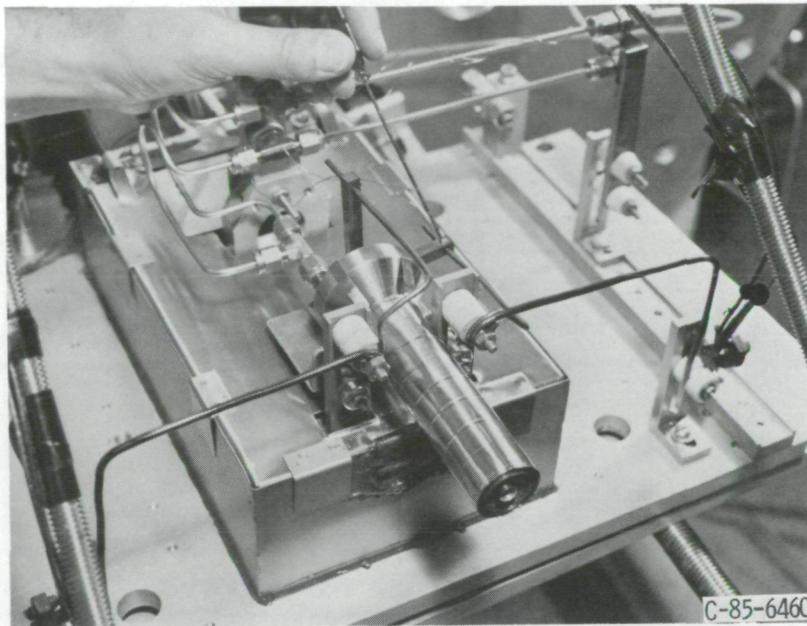
Figure 3. - Multipropellant resistojet mounted on thrust measurement system.
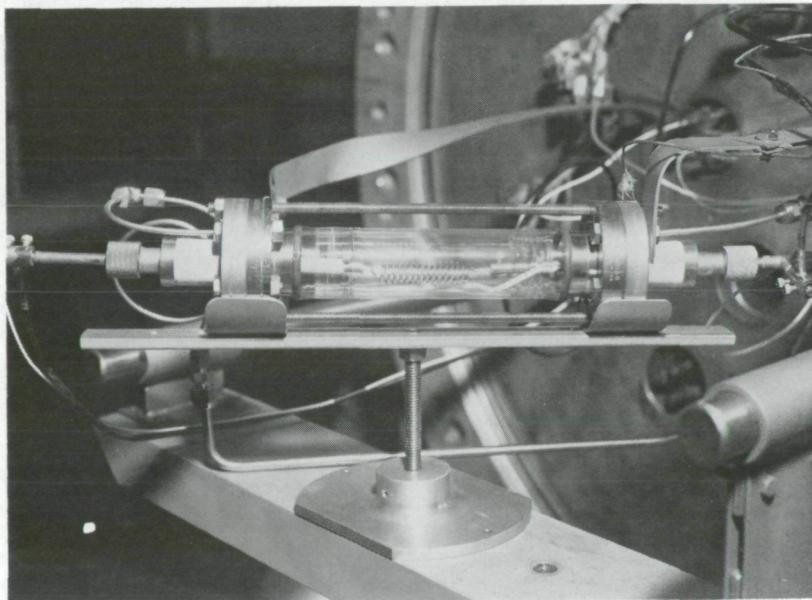
Figure 4. - Material test cell.
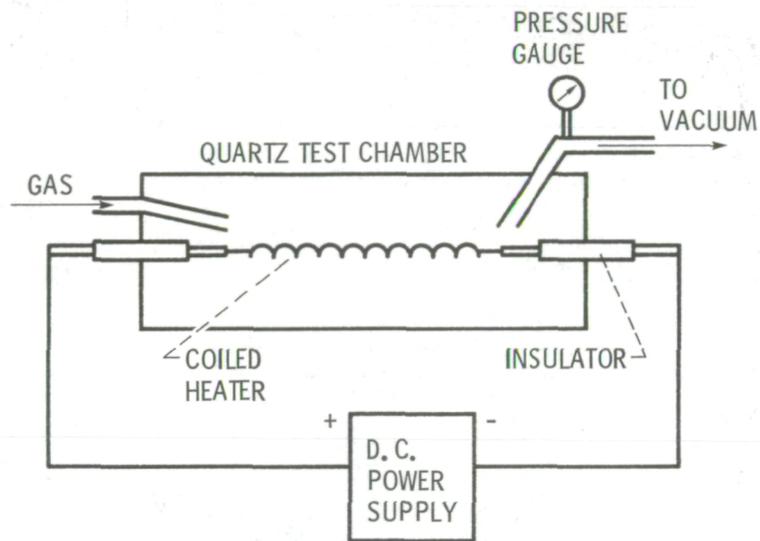
Figure 5. - Material experimental system.
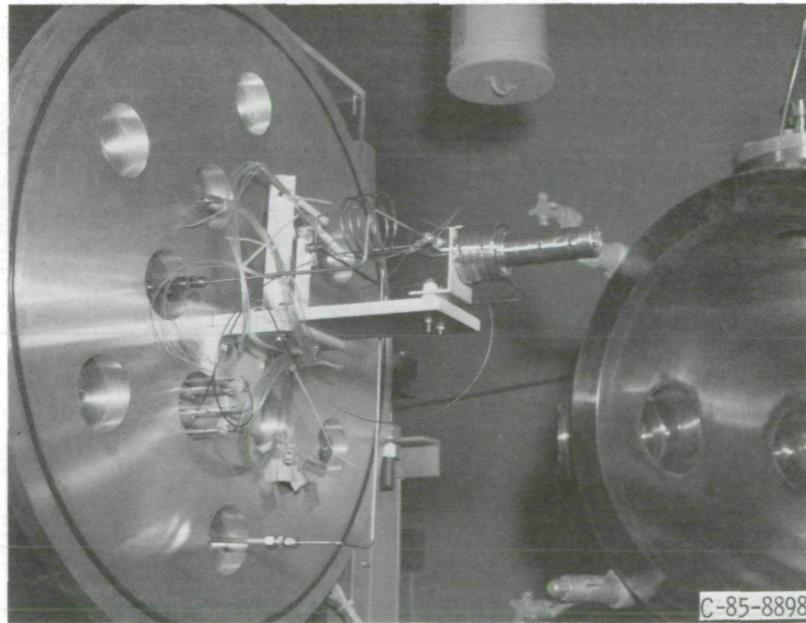
Figure 6. - Multipropellant resistojet installed in life test facility.
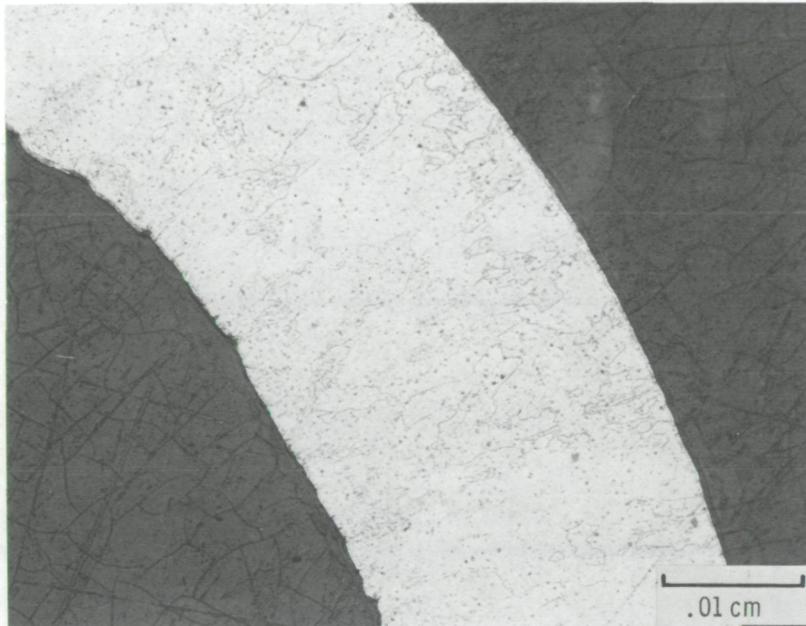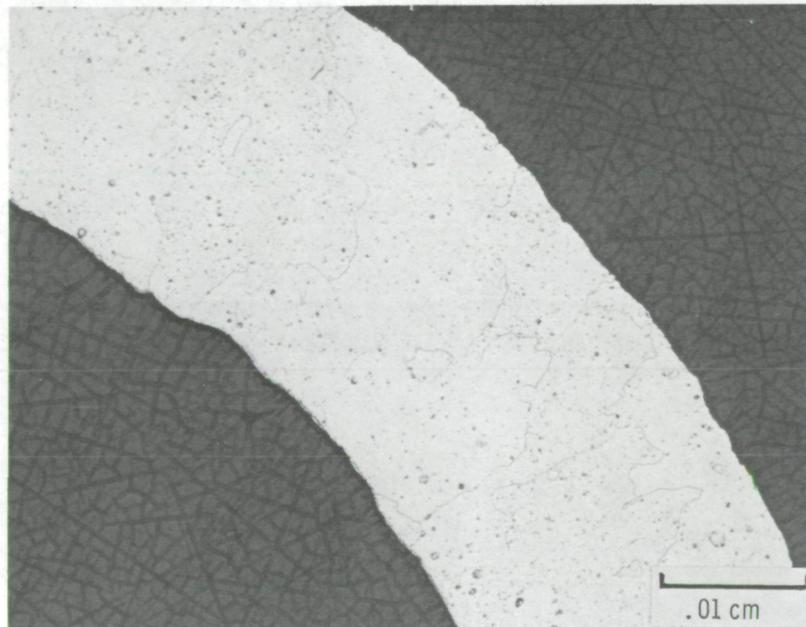
Before

After

Figure 7. - Cross section of Pt/Y_2O_3 tube before and after testing for 2000 hr at 1300 $^\circ$C in CO_2.
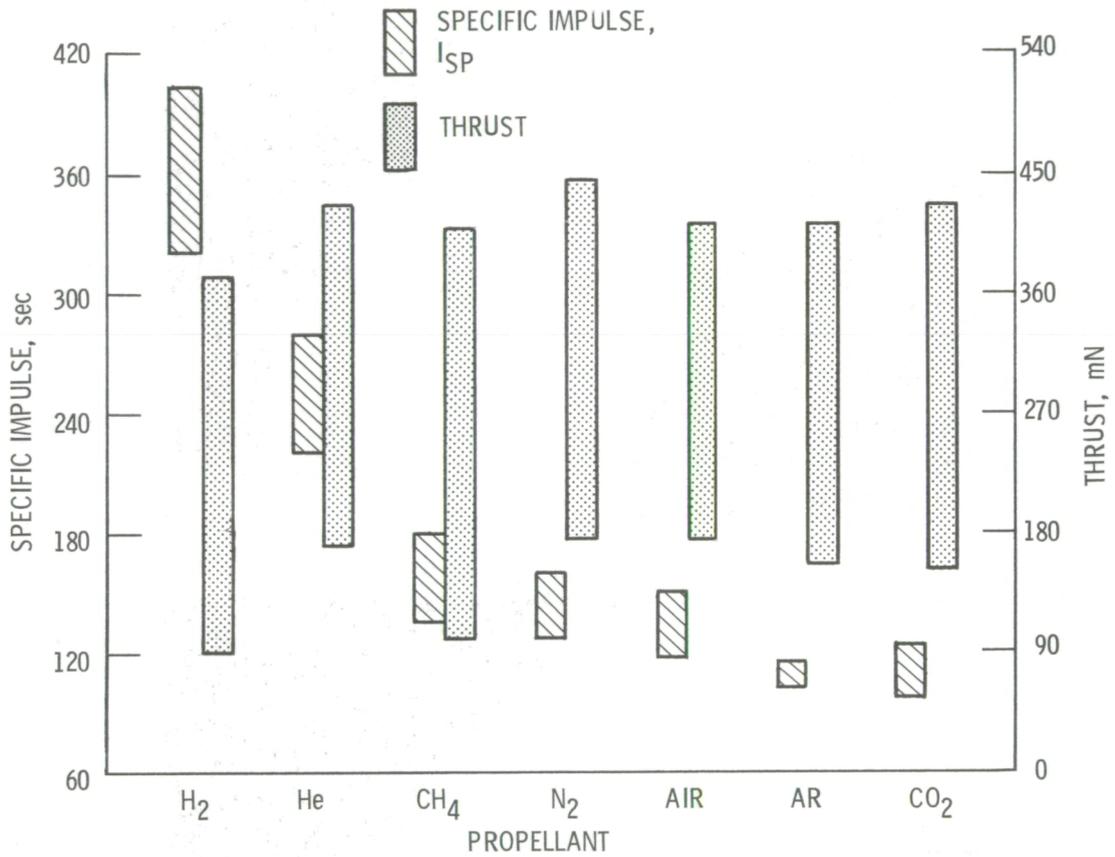
Figure 8. - Specific impulse and thrust ranges for a multipropellant resistojet operated on various propellants at a heater temperature of 1400 °C.
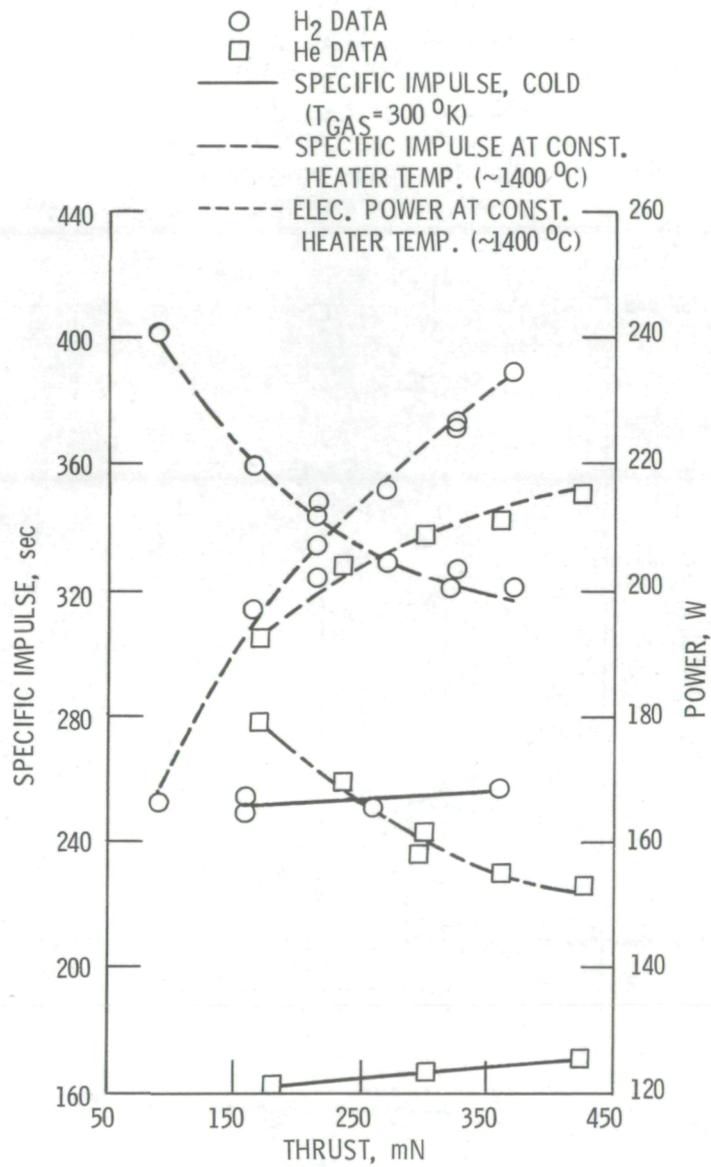
Figure 9. - Specific impulse and input power vs. thrust for multipropellant resistojet using H_2 and He.
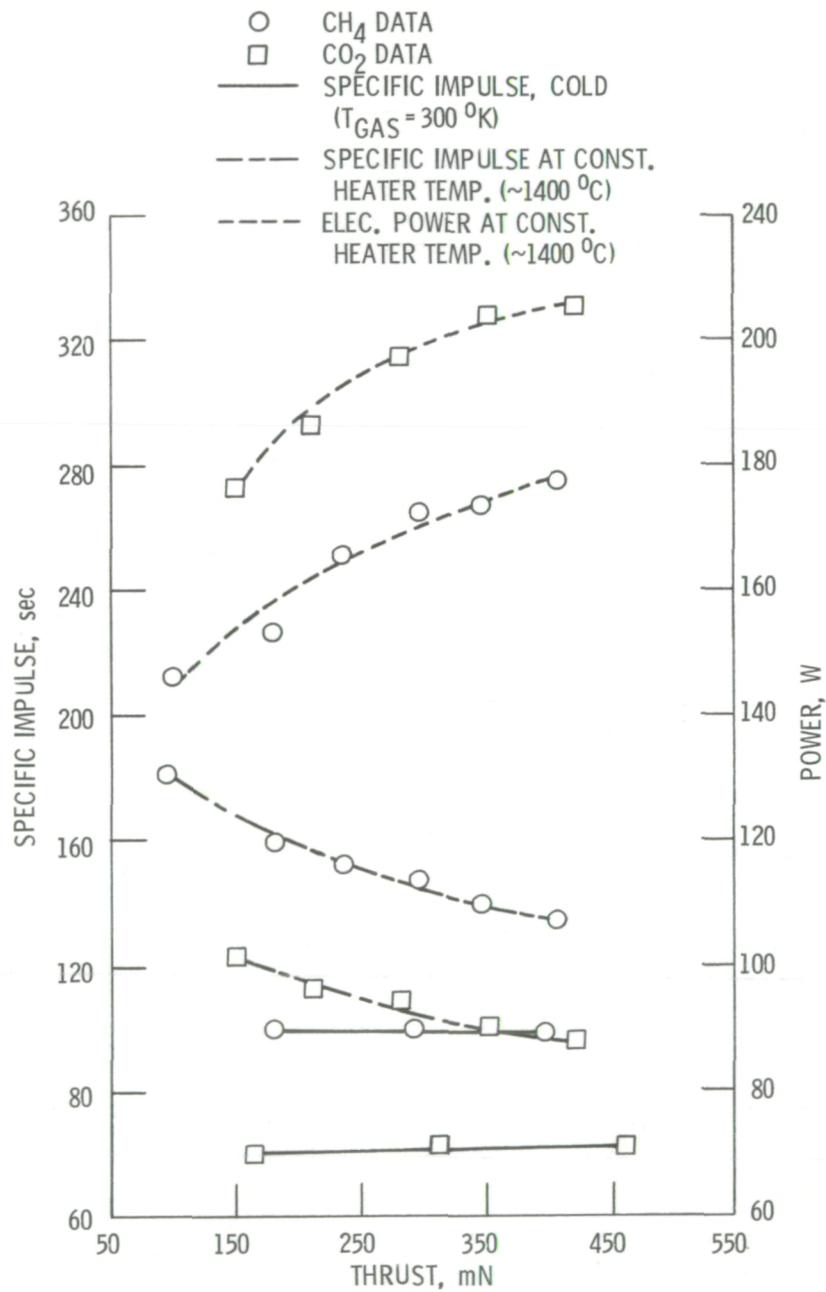
Figure 10. - Specific impulse and input power vs. thrust for multipropellant resistojet using CH_4 and CO_2.
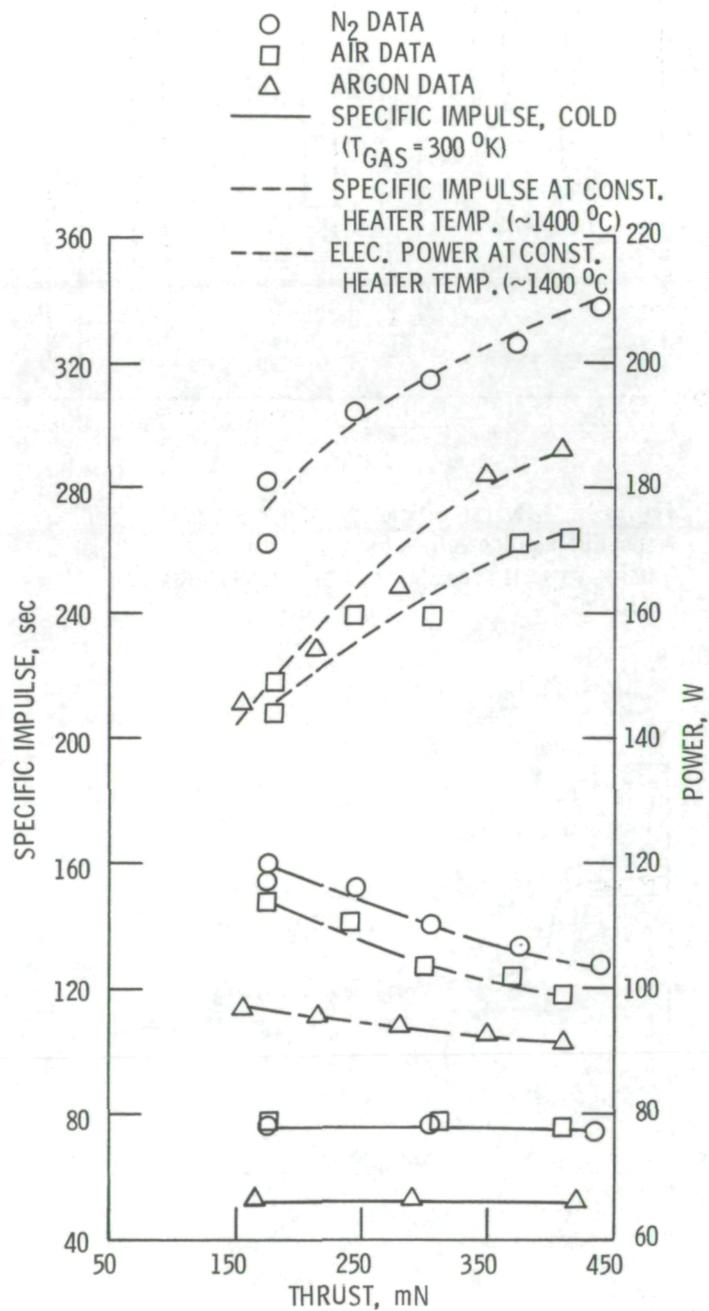
Figure 11. - Specific impulse and input power vs. thrust for multipropellant resistojet using N_2, air, and Ar.
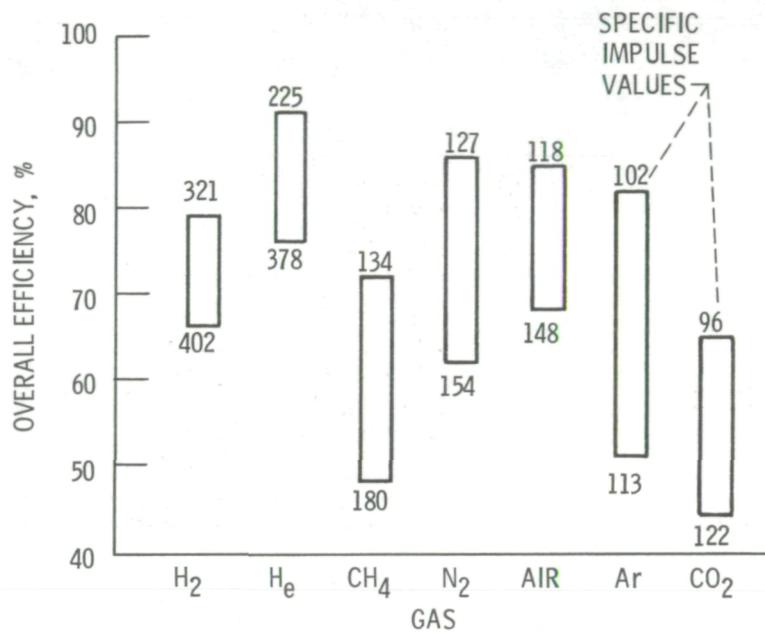
Figure 12. - Overall efficiency ranges and corresponding specific impulse values for multipropellant resistojet using various propellants at heater temperatures of 1400 $^{\circ}$C.
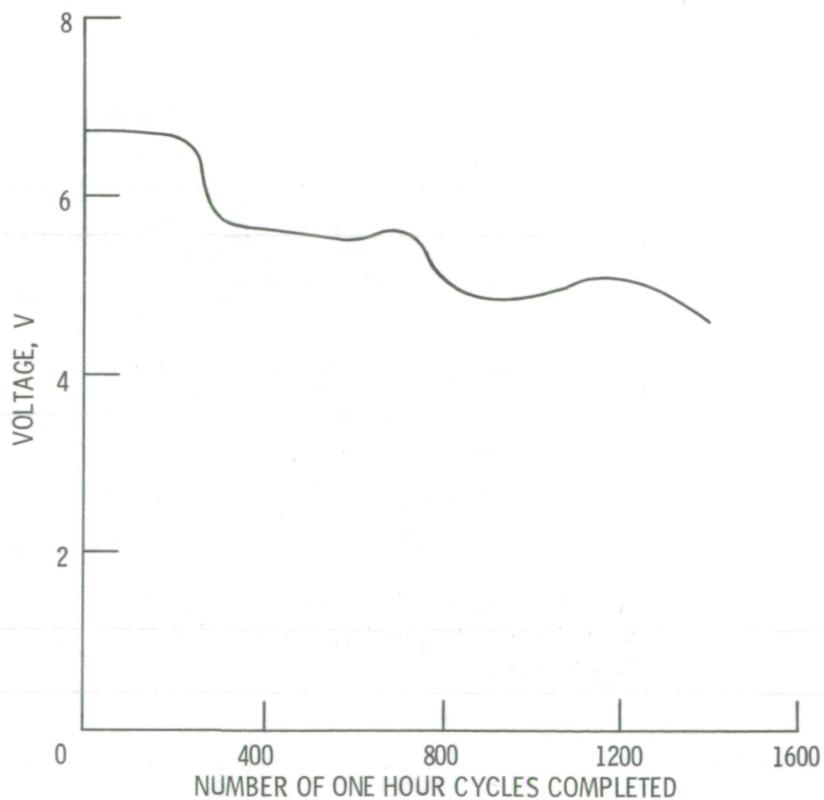
Figure 13. - Life test thruster heater voltage vs. time. 83% Heater duty cycle. 29A heater current.
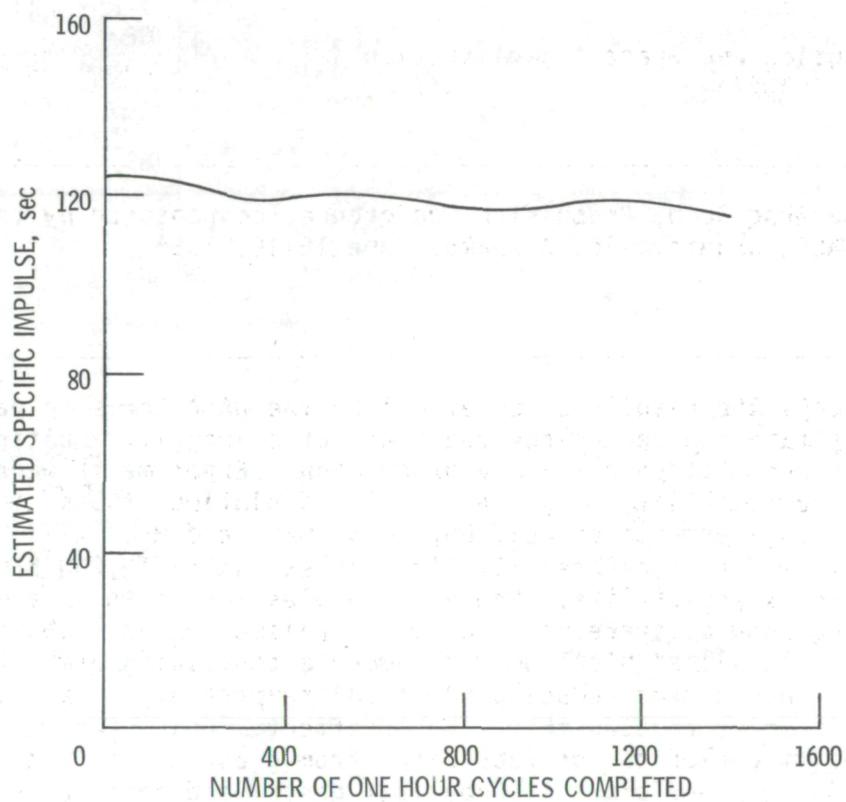
Figure 14. - Estimated specific impulse for life test multipropellant resistojet vs. time. 83% Heater duty cycle.

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16. Abstract

This paper presents the results of an effort by the NASA Lewis Research Center (LeRC) to demonstrate the technology readiness of a long-life multipropellant resistojet for Space Station auxiliary propulsion. Experiments were performed to evaluate the compatability of grain-stabilized platinum tubes at temperatures up to 1400 °C in environments of CO_2, CH_4, NH_3, H_2O, and H_2. All samples tested showed extrapolated lifetimes in excess of 10,000 hr based on 10 percent mass loss as end-of-life. However, samples tested in an ammonia atmosphere at 1400 °C showed severe pitting, which raised concerns about the compatability of grain-stabilized platinum with ammonia-containing atmospheres. Additional tests showed that reducing the metal temperature to about 900 °C (±100 °C) significantly reduced this adverse effect. Characterizations of a simple, radiatvely coupled design fabricated from grain-stabilized platinum on a variety of fluids were performed. Propellant gases used included hydrogen, helium, methane, air, nitrogen, argon, and carbon dioxide. Thrust levels observed ranged from 90 mN to 420 mN at input electric power levels from 140 to 240 W. Measured specific impulse values ranged from 100 to 400 sec. Endurance testing of the same type of thruster using CO_2 propellant was begun, and at the time of this writing 1400 operating cycles with on-time of about 1160 hr have been achieved.

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