H2 ARCJET PERFORMANCE MAPPING PROGRAM

Subcontract 90-1314/7293

FINAL REPORT

92-R-1615

Prepared for:

Texas Tech. University/NASA LeRC/SDIO

Submitted by:

Rocket Research Company
Olin Aerospace Division
11441 Willows Road N.E.
Redmond, WA 98073-9709

January 31, 1992

ROCKET RESEARCH COMPANY

REDMOND, WASHINGTON

Olin AEROSPACE DIVISION
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1.0 Introduction/Summary

This is the final report for subcontract 90-1314/7294 prepared for Texas Tech. University by Rocket Research Company. This work was performed during the period of March, 1991 to January, 1992.

High power H₂ arcjets are being considered for electric powered orbit transfer vehicles (EOTV). Mission analyses indicate that the overall arcjet thrust efficiency is very important since increasing the efficiency increases the thrust, and thereby reduces the total trip time for the same power. For example, increasing the thrust efficiency at the same specific impulse from 30% to 40% will reduce the trip time by 25%. For a 200 day mission, this equates to 50 days, which results in lower ground costs and less time during which the payload is dormant. Arcjet performance levels of 1200 seconds specific impulse (Isp) at 35% to 40% efficiency with lifetimes of over 1000 hours are needed to support EOTV missions.

The power level targeted for the present work was 10-15 kW. Such a thruster can be used to support EOTV's having power levels from 20 kW and higher by firing several thrusters simultaneously. Although higher efficiencies are generally obtained at higher powers, it was thought that this power level provided an appropriate balance between efficiency and near term mission suitability.

Little work has been done recently to optimize the H₂ arcjet performance. Work done in the 1960's by the Giannini Corporation focused on a unique arcjet configuration wherein the arc attaches in a subsonic region upstream of the throat, and which used regenerative cooling passages to improve the efficiency. This configuration is shown in Figure 1-1. Very high efficiencies were quoted for this configuration, and a 500 hour lifetest was completed at 1000...
GIANNINI ARCJET CONCEPT

Figure 1-1
seconds $I_{sp}$ and 55% efficiency at 30 kW input power. In theory, this configuration reduces the frozen flow losses by increasing the recombination in the subsonic constrictor. The regenerative passages both cool the constrictor region and pre-heat the incoming propellant. These high efficiency levels were not achieved during tests conducted at AVCO during the same period, nor has NASA or RRC come close with recent conventional arcjet tests.

Because of the potential to achieve very high efficiency levels, the objective of this program was to evaluate the ability of a scaled Giannini-style thruster to achieve the performance levels quoted in the previous literature while operating at a reduced nominal power of 10 kW. To meet this objective, a review of the past literature was conducted, scaling relationships were developed and applied to establish critical dimensions, a development thruster was designed with the aid of the plasma analysis model KARNAC and finite element thermal modelling, test hardware was fabricated, and a series of performance tests were conducted in RRC's Cell 11 vacuum chamber with its null-balance thrust stand.

The results obtained with the single configuration tested are encouraging. The thruster operated very stably over a power range of from 4 kW to 12 kW. There was virtually no erosion seen after approximately 20 hours of operation. Performance values were very repeatable. The efficiency levels obtained up to about 950 seconds $I_{sp}$ were significantly higher than for conventional designs. Above that $I_{sp}$ level, the efficiency was lower. These trends are shown in Figure 1-2 for 10 kW thruster power, along with RRC IRAD and NASA conventional data.

Although the initial results fall short of reproducing Giannini's reported performance levels, it is recommended that further investigations of this novel arcjet concept be conducted. Body temperature measurements and estimates of the effectiveness of the regenerative passages suggest that the efficiencies could be improved by reducing the thermal losses to the long subsonic constrictor. As will be discussed in more detail in the rest of the report, it appears that the constrictor was sized too large for this power level. Modifications to increase the temperature of the constrictor region are also recommended. Specific design recommendations are presented in section 7.0 Conclusions/Recommendations.
Isp Versus Efficiency
Hydrogen Propellant, 10kW

Figure 1-2
2.0 Review of Past Work

A review of the literature on the Giannini thruster shown in Figure 2-1 was completed. Several observations were made:

1. The thrust efficiencies reported for the radiation cooled design are 10 to 15 percentage points higher than for the AVCO R-2 design in the 1000 second specific impulse range, but drop below the AVCO data above 1200 seconds, as shown in Figure 2-2.

2. The regenerative design increases the efficiency by approximately 10 percentage points (to 55% nominal) at around 1000 seconds. This compares with 42% for the AVCO design. Higher specific impulse values could not be obtained with this configuration due to high thermal loads on the electrodes.

3. Operating voltages were very high, typically in the 200 to 250 V range, for the regenerative design.

4. The thermal design was very critical. Numerous mechanical difficulties were experienced with the regenerative design.

3.0 Design/Analysis Summary

A scaling exercise was conducted to produce a 10 kW design. There are four parameters that were used. These are: power/flow rate (P/m-specific energy); power/throat area (P/A*-power flux); inlet pressure (p_in); and the heat flux to the constrictor surfaces (Q_s/constrictor area). At constant thrust efficiency, P/m is proportional to Isp^2. The inlet pressure affects the wall heat transfer and the chemical kinetic rates. Chemistry rates for ionization and dissociation by two body collisional processes are proportional to p^2, while the reverse reactions are three body processes proportional to p^3. Maintaining the same inlet pressure during scaling should keep the pressure profile similar, and hence the heat flux and chemistry rates should also be similar.
H2 ARCJET DATA COMPARISON

- NASA H2, 10 kW
- GIANNINI H2 RAD, 25 kW
- GIANNINI H2 REGEN., 25 kW
- NASA H2, 30 kW, 0.100" Ø th.
- NASA H2, 30 kW, 0.070" Ø th.
- AVCO R-2, 30 kW

Figure 2-2
The throat and inlet pressures can be related to the heat added in the subsonic region of choked flow. The throat pressure can be estimated as:

$$p^* = \frac{1}{A^*k^*} \frac{1}{\sqrt{\frac{k^*+\frac{1}{2}}{m}}} \sqrt{P}$$

To scale with constant $p^*$ and constant $P/rh$ results in constant $P/A^*$. Heat flux considerations suggest that the lengths be scaled with radii. The results of these scaling methods are summarized in Figure 3-1.

An analysis was conducted using RRC's KARNAC code to calculate the energy flux to the constrictor walls and the current distribution. The results are shown in Figure 3-2. There is a very concentrated heat flux at the upstream corner of the constrictor. This is due primarily to the radiation from the high ohmic heating zone just off the cathode tip. There is another high energy flux region at the throat, as might be expected. This profile was incorporated into the thruster thermal model.

The current flux distribution shows most of the current being distributed along the slightly expanded region of the constrictor just before the throat. This supports the assumption that most of the heat addition is occurring in the subsonic region.

The thruster design is shown in Figure 3-3. The diameter of the anode is 2.0". Conax-style ram seals are used to seal the thruster around the cathode. The anode consists of two parts, which was necessary to create the stepped subsonic constrictor. One part includes the nozzle, throat, and part of the constrictor. It is lapped and press fit into a molybdenum body. The other part includes the injector holes and the upstream part of the constrictor. The joint between the two anode halves is lapped. The regenerative passage is created by forcing the gas down the outside of an annular baffle which is held in place between the injector and the cathode insulator. The gas returns along the outside wall of the constrictor, and then passes through the injector. The insulator parts are held in place by Belville washers which are trapped between the insulator tube and part of the Conax seal assembly.

The molybdenum anode body is attached to the main arcjet body by means of a large nut. Split rings are placed around the anode body after the nut is slipped over the upstream end. A graphoil seal is used between the two bodies. The tolerances around the seal are very tight to
### GIANNINI SCALING

![Diagram of GIANNINI SCALING](image)

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Principle Effect</th>
<th>Scaling Rational</th>
<th>30 kW</th>
<th>10 kW</th>
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<tbody>
<tr>
<td>$P/m$</td>
<td>$I_{sp}^2$ and enthalpy</td>
<td>Keep $I_{sp} = 1,000$ seconds</td>
<td>0.25 gm/sec</td>
<td>0.083 gm/sec</td>
</tr>
<tr>
<td>$D_3$</td>
<td>Controls choking point and subsonic heat addition</td>
<td>Keep $p^* = \text{const}$. $\Rightarrow P/D_3^2 = \text{const.}$</td>
<td>0.475 cm</td>
<td>0.274 cm</td>
</tr>
<tr>
<td>$D_1$</td>
<td>Velocity in subsonic section</td>
<td>$D_1/D_3 = \text{const.}$</td>
<td>0.635 cm</td>
<td>0.367 cm</td>
</tr>
<tr>
<td>$L_1$</td>
<td>Area of constrictor</td>
<td>Keep heat flux similar $L_1/D_1 = \text{const.}$</td>
<td>3.18 cm</td>
<td>1.83 cm</td>
</tr>
<tr>
<td>$D_2$</td>
<td>Control current attachment</td>
<td>$D_2/D_1 = \text{const.}$</td>
<td>0.699 cm</td>
<td>0.403 cm</td>
</tr>
<tr>
<td>$L_2$</td>
<td>Control current attachment</td>
<td>$L_1/L_2 = 2$</td>
<td>1.59 cm</td>
<td>0.915 cm</td>
</tr>
<tr>
<td>$\alpha_1$</td>
<td>Mack acceleration and recombination in nozzle</td>
<td>Adjust with $h^*$ or $I_{sp}^2$ or $P/m$</td>
<td>$30^\circ$</td>
<td>$30^\circ$</td>
</tr>
<tr>
<td>$\alpha_2$</td>
<td>Arc startup and stability</td>
<td>Past experience</td>
<td>$45^\circ$</td>
<td>$30^\circ$</td>
</tr>
</tbody>
</table>
ANALYSIS RESULTS – GIANNINI STYLE THRUSTER

ANODE HEAT LOADING

CURRENT FLUX DISTRIBUTION

Figure 3-2
minimize the exposure of the graphoil to the H\textsubscript{2} to prevent degradation of the seal. The position of this seal was determined on the basis of thermal model predictions to maintain the temperature below 1100°F. RRC's past experience indicated that the graphoil seal would not degrade at these temperatures when exposed to H\textsubscript{2}. An anti-seize compound is used to prevent the threads from galling. The propellant feed line is brazed into the side of the molybdenum arcjet body.

This thruster proved to be easy to assemble, and allowed for disassembly and reassembly while mounted on the thrust stand. No leaks occurred during any of the testing. The thruster is shown in Figures 3-4 and 3-5.

Originally a non-regenerative design was also created, as shown in Figure 3-6. This uses most of the same parts, with the exception of the two anode halves. The intent was to determine through testing the impact of the regenerative heating of the propellant. Unfortunately, funding limitations prevented this configuration from being built and tested.

Thermal modelling was performed to predict temperatures in key areas. The finite element model is shown in Figure 3-7. Several iterations were made before the final design was selected. The input conditions were an H\textsubscript{2} flow rate of 115 mg/sec, and a power distribution on the anode as predicted by the KARNAC analysis discussed above. The total power deposited along the constrictor walls was about 3200 W. Conduction and radiation to ambient temperature surroundings were assumed.

Originally, the gas was allowed to enter into an annular plenum area before passing through the injector holes. The model predicted that the gas was cooled significantly in this plenum, so it was eliminated.

Initial constrictor temperatures were excessive, so the anode nozzle wall thickness was increased to conduct more heat to the outer body.

An assessment was done of the effectiveness of the heat exchanger passage. The contact surface area was arbitrarily increased by a factor of two. This had little effect on the predicted gas temperature, so it was decided to not add any tortuous paths (e.g., a helical flow path) to the heat exchanger.
Figure 3-4
Runs were made with both a natural exterior with an emissivity of 0.2 and a spray coated exterior with an emissivity of 0.6. None of the critical temperatures were exceeded in either case, and for the 0.2 emissivity case, the gas temperature was increased by 200°F. As a result, it was decided to not coat the exterior.

The final predicted temperatures are shown in Figure 3-8. It should be noted that the greatest uncertainty in the model predictions is a result of the inability to precisely know how much power is input to the thruster structure during firing. RRC has correlated thermal model predictions with measured results on low power arcjets, and has achieved excellent agreement after several iterations. However, this kind of iterative process to refine the thermal model was beyond the scope of this program. This should be considered for future work so that the model predictions can be used to optimize the design. In this program, the model results were used to target key potential problem areas, but were not intended to provide exact temperature predictions.

4.0 Test Facilities/Procedures

Testing was performed in Cell 11 of the RRC Electric Propulsion Test Facility. A vacuum environment for arcjet operation is established by either one or two Stokes Model 1729 mechanical pumps. A plot of vacuum level as a function of hydrogen mass flow is shown in Figure 4-1.

Electrical power to the arcjet was supplied with a Hypertherm MAX 100 Plasma Cutting DC Power Supply. The output power level and on/off was controlled remotely from the control room. As shown in Figure 4-2, the electrical power circuit includes additional components used for the start up procedure. A capacitive high voltage start circuit is located in parallel with the DC output power leads to generate the voltage levels required for arc breakdown between the electrodes.

The hydrogen propellant was of Grade 4.7 (99.997 % purity) contained in industrial type high pressure cylinders. A pressure regulator located at the outlet of the cylinder was used to maintain an adequate pressure level into the system. A Micro-Motion Model D-6 was used as the mass flow meter. Remote flow control was accomplished by adjusting an electric motor attached to the shaft of a pressure regulator. A schematic of the propellant system is shown in Figure 4-3.
Final Thermal Model Predictions

Thickness increased to block gas flow

Natural Exterior

$m = 115 \times 10^{-6} \text{ Kg/sec} \quad \text{Hydrogen}$

Wall thickness substantially increased
Vacuum Level vrs H₂ Mass Flow

FIGURE 4-1
ELECTRICAL POWER DELIVERY SCHEMATIC

Figure 4-2
Figure 4.3

GIANNINI ARCJET PROPELLANT SCHEMATIC

HAND VALVES
HV1: H2 BOTTLE VALVE
HV3: FLOW METER ISOLATION
HV4: FLOW METER ISOLATION
HV5: BACKFILL PURGE ISOLATION
HV6: PROPELLANT ISOLATION
HV7: GN2 ISOLATION
HV8: BLEED OUTLET VALVE

SOLENOID VALVES
SV1: H2 SUPPLY ISOLATION (N.C.)
SV2: BACKFILL PURGE ISOLATION (N.C.)
SV2: OFF PULSE VALVE (N.O.)
SV4: BLEED VALVE (N.C.)

CHECK VALVES
CV2: H2 INLET

PRESSURE REGULATORS
R1: BACKFILL PURGE
R2: FEED PRESSURE
R3: MOTORIZED FLOW CONTROL

PRESSURE GAUGES
G1: BOTTLE PRESSURE
G2: BACKFILL PURGE PRESSURE
G3: R3 FEED PRESSURE
G4: D2 BOTTLE PRESSURE

HV5
SV2
SV3
3" Nylon Isolator
P(Inlet) Pressure Transducer

HV8
HV4
HV3
HV6
Filter
Micro-Motion Flow Meter

R1
G1
G2
G3
R2
R3

CV2
H2

[Diagram of propellant schematic with valves and flow control system]
Data were continuously displayed and were also recorded at timed intervals on a PC based computer controller system. Strip chart recorders were used to record signals that require a graphical display to monitor transients and noise levels. The instrumentation list is shown in Table 4-1.

**Table 4-1**  
Instrumentation List

<table>
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<th>Parameter</th>
<th>Range</th>
<th>Accuracy</th>
<th>Device</th>
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<tr>
<td>arc voltage</td>
<td>0 - 400 vdc</td>
<td>+/- 1.0%</td>
<td>1000:1 resistive voltage divider</td>
</tr>
<tr>
<td>arc current</td>
<td>0 - 100 amps</td>
<td>+/- 1.0%</td>
<td>300 amp resistive current shunt</td>
</tr>
<tr>
<td>thrust</td>
<td>0 - 2.0 N</td>
<td>null balance</td>
<td>+/- 1.5%</td>
</tr>
<tr>
<td>propellant mass flow</td>
<td>0 - 200 mg/s</td>
<td>+/- 1.0%</td>
<td>Micro-Motion Model D-6</td>
</tr>
<tr>
<td>arcjet inlet pressure</td>
<td>0 - 200 psia</td>
<td>+/- 0.5%</td>
<td>Statham pressure transducer</td>
</tr>
<tr>
<td>vacuum tank pressure</td>
<td>0 - 1 Torr</td>
<td>+/- 5%</td>
<td>MKS Baratron vacuum gauge</td>
</tr>
<tr>
<td>arcjet temperatures</td>
<td>2300 F max.</td>
<td>+/- 0.75%</td>
<td>Type K thermocouples</td>
</tr>
</tbody>
</table>

Prior to the first augmented firing of the arcjet and at times throughout the test firings, several cold H₂ flow sequences were performed. Plots of cold flow thrust, inlet pressure, and mass flow were generated and maintained throughout testing as an indication of potential changes in the nozzle geometry or of the development of gas leaks. The cold flow tests were performed with thruster temperatures below 200°F to minimize the effect of gas heating on the performance parameters.

Prior to operating the arcjet with the DC power supply, a test is conducted at nominal mass flow rates which verifies that arc breakdown is occurring between the electrodes and not elsewhere in the electrical circuit. An arc discharge between the electrodes is generated by shutting off the mass flow for approximately 100 msec, which lowers the pressure within the nozzle enough to
create a Paschen breakdown. During this checkout procedure only the capacitive start circuit is used to minimize the potential for damage in case of an inadvertent breakdown.

Arcjet startup was accomplished by turning on the DC power supply at its maximum open circuit voltage level and creating a discharge as mentioned above. The capacitors sustain the arc until the DC power supply responds to the load and starts conducting and controlling current. Arc voltage and current start transient data were measured with a Tektronix P6015 broadband high voltage probe and a Tektronix A6303 current probe. Both signals were recorded with a Tektronix 2431L digital storage oscilloscope.

Power and mass flow rate are adjusted to obtain the desired power/mass flow ratio. Thermal equilibrium of the thruster typically requires 30 minutes of operation and is determined by monitoring temperature readings. Steady state performance measurements are recorded by averaging 300 samples, where each sample represents the average of 30 readings from the analog/digital board of the PC based controller. The time period for the recording process is roughly 10 seconds. Because of thermal zero shift of the thrust stand, performance measurements are obtained by using post-firing zeros as opposed to pre-firing zeros. Immediately after recording performance data at equilibrium conditions, the arc power is shut off, then the propellant valve located near the arcjet inlet is closed. Ten seconds elapse to allow residual gas to exit the arcjet nozzle prior to recording the post-firing zero thrust, inlet pressure, and mass flow rate measurements. The corrected engineering data are both recorded on a hard disk and printed.

5.0 Test Results

Performance data were taken over a range of power levels from 4 kW to 15 kW. Figure 5-1 shows I<sub>sp</sub> vs. thrust efficiency over this range of power. Test firings at 4 kW to 6 kW resulted in thrust efficiencies greater than 50% and exhibited low anode surface temperatures, no visible exhaust plume, and stable operation. Increasing the power level at a fixed P/r<sub>m</sub> level resulted in only slight improvements in efficiency. For example, increasing the power from 8 kW to 12 kW at 60 MJ/kg specific power caused the efficiency to increase from 49.5% to 51.2%. Figure 5-2 shows the 10 kW data taken along with data from a standard constricted design tested by RRC on another program. At specific impulse levels below 950 sec the Giannini style thruster produced much higher efficiencies. However, above this performance level, the standard design was more efficient.
Isp Versus Efficiency
Hydrogen Propellant

Figure 5-1
Isp Versus Efficiency
Hydrogen Propellant, 10kW

Figure 5-2
Figure 5-3 shows anode surface temperatures over a range of powers from 8 kW to 12 kW at a relatively low P/m level of 60 MJ/kg. The temperatures drop between 250°F and 400°F as the power and flow rate are increased. Although the heat loading is greater at 12 kW, the additional cooling by the increased mass flow results in lower temperatures. Figure 5-4 shows temperatures for a fixed power of 10 kW over a range of P/m levels. The temperatures climbed by close to 1000°F at the nozzle end as the P/m level was increased from 60 MJ/kg to 80 MJ/kg.

Voltage vs. current traces are shown in Figure 5-5. The range of voltages was between 120 V and 180 V. Figure 5-6 shows the arc voltage vs. flow rate. Apparently the arc voltage is directly dependent on the flow rate, and is not strongly dependent on the power level. Figure 5-7 shows the measured inlet pressure external to the thruster vs. mass flow rate.

A complete set of data from these tests is included in the appendix.

A boroscope was used periodically during testing to inspect the constrictor. This was recorded on a video tape. The surface finish was rougher just downstream of the small step in the constrictor, suggesting that this is where the arc was attaching. No erosion was observed. The throat region was recrystallized. Post-test inspection of the disassembled hardware indicated no difference from the boroscope findings.

6.0 Discussion

The main issue based on the test data is to determine why the efficiency drops off so rapidly as the P/m level is increased. Two approaches were taken to attempt to better understand this behavior.

During testing, the thrust level was recorded just after the arc was extinguished but with the flow still on. Using this and the end-of-run corrected total thrust, a measure of the amount of heat picked up from the structure by the gas can be obtained. The thrust measured just after the arc is shut off is in essence due to the resistojet-like behavior of the thruster. This thrust level can be correlated with an enthalpy level using the CEC chemical kinetics program. This provides an estimate of the total power being absorbed by the gas from the flow passages by multiplying the enthalpy by the mass flow rate.
Anode Temperature Profile

Specific Power = 60 MJ/kg

- Increasing mass flow and power yields increased efficiency combined with decreasing surface temperatures.

Figure 5-3
Anode Temperature Profile

Power = 10 kW

![Graph showing Anode Temperature Profile with three curves for different MJ/kg values: 60, 70, and 80 MJ/kg. The x-axis represents Distance Along Anode (inches), and the y-axis represents Anode Surface Temperature (Degrees F). The graph includes markers for each MJ/kg value.]
Arc Voltage versus Arc Current
Hydrogen Propellant

Figure 5-5
Arc Voltage versus Mass Flow Rate
Hydrogen Propellant

Figure 5-6
Inlet Pressure versus Mass Flow Rate
Hydrogen Propellant

Inlet Pressure (psia)

Mass Flow rate (mg/s)

Figure 5-7
Figure 6-1 shows the predicted $I_{sp}$ as a function of enthalpy based on CEC model runs, assuming that the flow is in equilibrium upstream of the throat and frozen downstream of the throat. Subtracting a small amount of enthalpy contained in the ambient temperature inlet gas provides a total enthalpy increase. Multiplying by the flow rate gives the total power picked up by the gas. Table 6-1 shows the measured $I_{sp}$ levels just after the arc was shut off, and the calculated values of delta enthalpy and power. The estimated gas temperature upstream of the throat is also included. Figure 6-2 shows the arc-off $I_{sp}$ vs. specific power. The calculated power is graphed against $P/m$ in Figure 6-3.

Several observations can be made based on this approach. First, the estimated gas temperatures are in a range of 1400°F to 2450°F. External temperatures were measured to over 2500°F, and thermal modeling predicts that the temperature difference between the outer wall and the constrictor should be 1000-2000°F. In addition, pyrometer measurements taken looking up the nozzle indicated temperatures in excess of 3500°F at the high specific power levels. This evidence suggests that the constrictor walls are substantially hotter than the gas after the arc is shut off, which means that there is room for improvement in the effectiveness of the heat exchanger. This will be addressed in the recommendations section.

A second observation is that there is a distinct maximum in the power absorbed by the gas as a function of $P/m$. The gas temperature increases as the $P/m$ level is raised. However, above a $P/m$ level of about 80 MJ/kg, the drop in flow rate results in a reduced total power being absorbed by the gas. As a result, the increased thermal losses that occur at higher $P/m$ levels are compounded by the fact that less of the available energy can be recovered by the gas through the regenerative passages.

A second approach taken to better understand the rapid decrease in efficiency as $P/m$ was increased was to estimate the radiation losses from the thruster body. This was done by dividing the body into 3 distinct regions, and selecting an average temperature for each region based on thermocouple measurements. Data were not available for all operating points because at higher $P/m$ levels, the temperatures exceeded the limits of the type K thermocouples. Nonetheless, estimates were made for 5 of the operating points.

A key variable is the emissivity. The hemispherical total emittance of molybdenum increases with temperature. At 1340°F (1000 K), the emittance ranges from 0.10 to 0.15 for polished molybdenum. At 2420°F (1600 K), the emittance ranges from 0.15 to 0.20. Since the exact
Figure 6-1

PREDICTED SPECIFIC IMPULSE vs ENTHALPY

- NASA CEC PROGRAM
- $H_2$
- $\epsilon = 250$
- EQUILIBRIUM UPSTREAM OF THROAT, FROZEN IN NOZZLE

(seconds)$^{d_{s1}}$

ENTHALPY (J/kg x 10^6)
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<th>η Arc-On (%)</th>
<th>P/ṁ (MJ/kg)</th>
<th>Isp Arc-Off (sec)</th>
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<th>Enth CEC (MJ/kg)</th>
<th>Δ Enth (T₀ = 70°F) (MJ/kg)</th>
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Arc Off Isp versus Specific Power
Hydrogen Propellant, 10 kW

Figure 6-2
Figure 6-3

REGENERATIVE GAS POWER vs SPECIFIC POWER

GAS POWER (W)

P/m (Mg x 10^6)

2000 1900 1800 1700 1600 1500

BASED ON THRUST MEASURED IMMEDIATELY AFTER ARC SHUT-OFF
value was not determined experimentally, the calculations were performed over a range of values.

Figure 6-4 shows the locations of the thermocouples. The temperature data are included in the appendix. Table 6-2 shows the calculated radiation losses for assumed emittances of 0.15 and 0.20. A plot of radiative power vs. $P/\dot{m}$ is shown in Figure 6-5. It is apparent that these losses increase rapidly with increasing specific power. These calculations did not consider the heat conducted through the thruster, and are therefore a conservative estimate of the losses.

| Table 6-2 |
| Radiation Loss Estimates, 10 kW |

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<th>Data Point #</th>
<th>$P/\dot{m}$ (MJ/kg)</th>
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If the thruster efficiency drops from 40% to 30% at the same input power, this implies that 10% of the input power is no longer available as thrust. This energy could end up as frozen flow, or as thermal losses. It is not likely that such a large change could be due primarily to a decrease in nozzle efficiency. If all of it theoretically ended up as a thermal loss, this would define an upper bound for the slope of the thermal loss calculations. In other words, if at 10 kW the thruster lost 10 percentage points in efficiency, the maximum increase in the thermal losses would be 1.0 kW. Figure 6-6 shows the calculated radiative losses vs. thruster efficiency for the 5 points. Also shown are theoretical curves for both 100% and 50% of any additional losses being due to structural heating. From this graph it can be seen that a significant portion of the drop in efficiency must be due to increased thermal losses, even if the assumptions regarding the emissivity are not accurate. In the future, a shell model of the thruster could be constructed which would much more accurately calculate the radiative and conductive thermal losses.
Figure 6-5

ESTIMATED RADIATION LOSSES vs SPECIFIC POWER

\[ \epsilon = \text{EMISSIVITY} \]

\[ \epsilon = 0.20 \]
\[ \epsilon = 0.15 \]

\[ P_{\text{th}} \text{ (J/kg x 10^6)} \]

\[ \text{RADIATION (W)} \]

39
ESTIMATED RADIATION LOSSES vs EFFICIENCY
10 kW

$\epsilon = \text{EMISSIVITY}$

PREDICTED RADIATION LOSSES

$\epsilon = 0.20$

$\epsilon = 0.15$

THEORETICAL MAXIMUM $\frac{d(Q_{rad})}{d(Eff)}$ IF 100%

OF EFFICIENCY DECREASE WAS DUE TO
RADIATIVE THERMAL LOSSES FOR 10kW
INPUT POWER

THEORETICAL MAXIMUM $\frac{d(Q_{rad})}{d(Eff)}$ IF 50%

OF EFFICIENCY DECREASE WAS DUE TO
RADIATIVE THERMAL LOSSES FOR 10kW
INPUT POWER
The point of this discussion is that design changes are needed to reduce the thermal losses at higher \( P/r_n \) levels. Combining the gas power and radiative loss estimates as a measure of the total power entering the structure shows that out of 10 kW input power, well over 3000 W are being deposited to the thruster walls at \( P/r_n \) levels above 100 MJ/kg. Although some of this is recovered by the gas, the ratio of gas power to radiative losses rapidly decreases as the specific power is increased. Some of the design recommendations discussed below are aimed at reducing these losses.

### 7.0 Conclusions/Recommendations

Testing of a scaled Giannini-style thruster was completed successfully, with data taken over a range of power levels from 4 kW to 15 kW. The thruster ran very stably, and no erosion was evident. Efficiencies much higher than conventional designs were achieved at specific impulse levels up to 950 sec. Unfortunately, operation at specific impulse levels of interest (>1000 sec) was not achieved because of a rapid decrease in efficiency as the specific power was increased. This was apparently due to a very rapid increase in the thermal losses.

There are several recommendations for future work. First, steps should be taken to reduce the thermal losses. Reducing the length and diameter of the constrictor/throat would lower the surface area which is in contact with the hot gases. Second, there is room for improving the heat exchanger efficiency. This could be done by reducing the anode nozzle wall thickness to raise the constrictor temperatures, and by improving the gas path to increase the surface area and resident time. Third, RRC recommends that a non-regenerative configuration, as was shown in Figure 3-6, be built and tested to investigate separately the effect of the subsonic constrictor. Fourth, a shell thermal model of the thruster should be constructed to allow more accurate calculations of thermal losses based on thermocouple and CID camera measurements. This would provide a way to measure the degree to which design changes have affected the thermal losses. Last, the dependence on power should be evaluated further by testing at higher powers of up to 20 kW.

### References

APPENDIX

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