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MONOPROPELLANT HYDRAZINE RESISTOJET

PRELIMINARY DESIGN TASK SUMMARY REPORT

TRW SYSTEMS GROUP
ONE SPACE PARK
REDONDO BEACH, CALIFORNIA 90278

FEBRUARY 1972

PREPARED FOR
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GREENBELT, MARYLAND 20771

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SYSTEMS GROUP

FOREWORD

This is the second Task Summary Report submitted under Contract NAS5-11477, "Design, Development, and Testing of an Engineering Model 20-Millipound Thrust Monopropellant Hydrazine Resistojet." The program originated in the Auxiliary Propulsion Branch of the NASA/Goddard Space Flight Center. Mr. Dennis Asato is the Technical Officer for NASA/GSFC. Mr. Charles K. Murch is the Project Manager for TRW Systems Group. This document was prepared by Charles K. Murch.

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1. SUMMARY

This document summarizes the design effort in support of the "Design, Development, and Testing of an Engineering Model Nominal 20-Millipound Thrust Monopropellant Hydrazine Resistojet Program," NASA Contract NAS5-11477. The thruster, herein called the Electrothermal Hydrazine Thruster (EHT), decomposes hydrazine thermally and expands the decomposition products through a nozzle to provide the impulse necessary to fulfill spacecraft propulsive requirements. The thruster is capable of operation at pulse widths from 0.050 second to steady state and delivers specific impulse values from 175 to greater than 225 seconds, depending upon the duty cycle.

The design of the EHT is the result of an analytical effort[1] combined with a series of design verification tests. The design requirements, design philosophy and detailed design methods are outlined in this report. Also included is a discussion of the rationale behind the selection of materials for the EHT. Finally, the results of the design verification tests are presented.

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[1] Monopropellant Hydrazine Resistojet, Analysis Task Summary Report, TRW Document 20266-6006-R0-00, November 1971.

2. DESIGN

2.1 Design Requirements and Objectives

The thruster design and performance requirements specified by Goddard Space Flight Center are listed below.

a) Thrust: 20 ± 2 millipounds steady state at 100 psia feed system pressure

b) Vacuum Specific Impulse: 200 seconds minimum steady state
175 seconds minimum pulse mode

c) Pulse Duration: 0.050 second to steady state

d) Pulse Mode Duty Cycle: To be specified at first joint meeting at GSFC

e) Power to Thruster Assembly and Pressure Regulation Device: 15 watts maximum

f) Holding Power: 5 watts maximum

g) Impulse Bit Reproducibility: For any given feed system pressure and valve opening time, the delivered impulse bit shall be reproducible within ± 10 percent.

h) Weight: Not specified

i) Size: Not specified

j) Nominal Voltage: The thruster assembly, pressure regulating device (if required) and power conditioning module shall operate within the requirements herein at 24 to 32 Vdc.

k) Total Number of Pulses: 1 million

l) Maximum Steady State on Time: 30 hours.

For the purpose of testing the EHT, pulse mode duty cycles (Item (d) above) of 0.01, 0.02, 0.05, 0.1, 0.25, and 1.0 were selected. The thruster is designed for operation with no duty cycle limitations. Note also that the "Pressure Regulation Device" mentioned in the requirements is not applicable to the EHT design currently under development. Accordingly, the 15 watt requirement in Item (e) is superseded by the 5 watt requirement of Item (f).

In addition to the above requirements, TRW established a number of design goals and constraints. It was believed that accomplishment of these goals would assure future flight utilization.

- The thruster should be entirely compatible with the propellant supply systems of existing, higher thrust catalytic hydrazine thrusters. This implies that the EHT should be capable of operation over a blowdown pressure range starting at several hundred psia.
- The pulsed mode and steady state specific impulse should be greater than that characteristic of catalytic engines operating at higher thrust levels. A steady state specific impulse goal of 230 seconds was established.
- The response times of the thruster should be well matched to the requirements of typical spacecraft control systems.
- The inherent reliability of the thruster should be at least as high as that of larger catalytic thrusters. Elimination of the catalyst bed increases the reliability but the dependence on heater power is a negative factor. For this reason, heater reliability is of prime concern. This concern dictated the separation of the heater from direct propellant exposure.

2.2 Description of Preliminary Design

Figures 2-1 and 2-2 indicate the preliminary design of the Electrothermal Hydrazine Thruster. Two different valve configurations have been tested to date; the Wright Components, Inc. Model No. 15650 and the Parker Model No. 5686070. A third solenoid valve configuration, Parker No. 5100060, will be used later in the program. A new assembly drawing will be provided for this configuration. The thruster/valve assembly in all cases has been designed to accept a single bracket attached to the solenoid valve. The thruster insulation is molded from WRP-X-AQ, a structurally-sound, rigid, machineable material. The chamber is heated by a 5-watt tubular resistance element capable of maintaining the thruster at about 1100°F during periods of no propellant flow. The nominal thrust level is 0.020 pound at 100 psia inlet pressure.

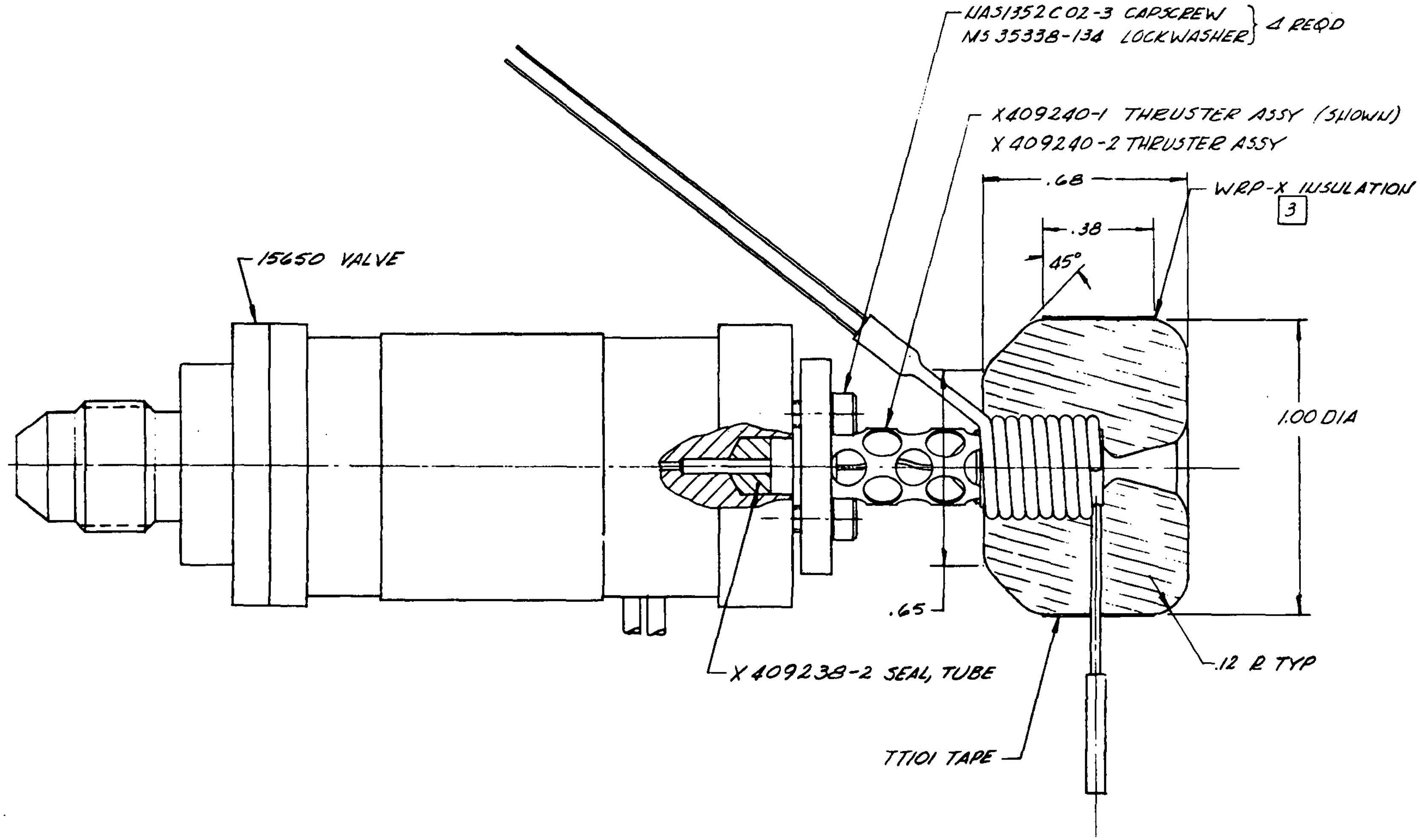


Figure 2-1. Preliminary Electrothermal Hydrazine Thruster Design

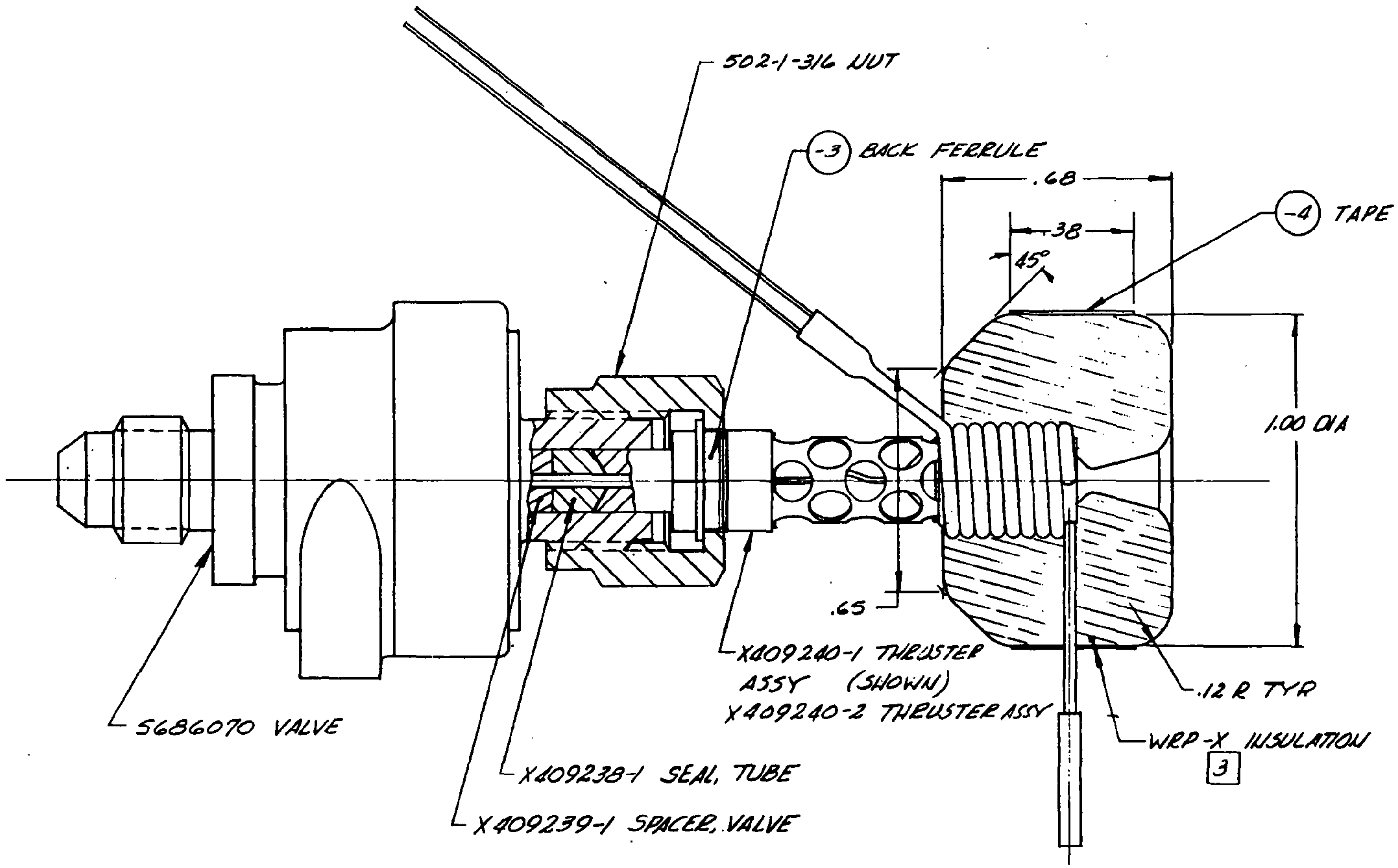


Figure 2-2. Preliminary Electrothermal Hydrazine Thruster Design

Figure 2-3 indicates the internal geometry of the thruster. The internal geometry is also indicated by Figure 2-4 which shows the assembly sequence of a test unit similar to the design shown in Figures 2-1 and 2-3.

Figure 2-4a shows the thruster prior to the initial braze operation. The nozzle is of standard convergent-divergent design with an area ratio of 50. The screen pack is cylindrical in shape and is formed from Haynes 25 mesh material. The chamber head end has a 90° included angle and a tapered wall thickness to limit heat transfer to the injector. The bolt flange adapts the entire assembly to the solenoid valve.

Figure 2-4b shows the thruster after the initial braze. A single heater element has been coiled around the chamber and a chamber pressure tap has been installed. The wire protruding from the inlet prevents contamination from entering the injector during handling and brazing.

Figure 2-4c indicates the completed assembly prior to installation of the insulation. The chamber, nozzle, and injector have been instrumented with thermocouples and a chamber pressure transducer has been installed. A large thermocouple-type termination has been installed on the heater for test purposes. The valve shown in the figure is a Wright Components unit. The injector-to-valve seal is accomplished with metallic spacers and a Teflon compression seal.

2.3 Mass and Energy Kinetic Design

Experience with thermal decomposition hydrazine thrusters has shown that the most important design factors to consider are: 1) the injection technique, and 2) the heat transfer, mass transfer, and fluid flwo processes which occur within the thrust chamber. These factors are not always totally amenable to analysis. Therefore, thruster design has been a combination of analytical and empirical efforts.

The sequence of processes which occur in the electrothermal thruster may be described as follows:

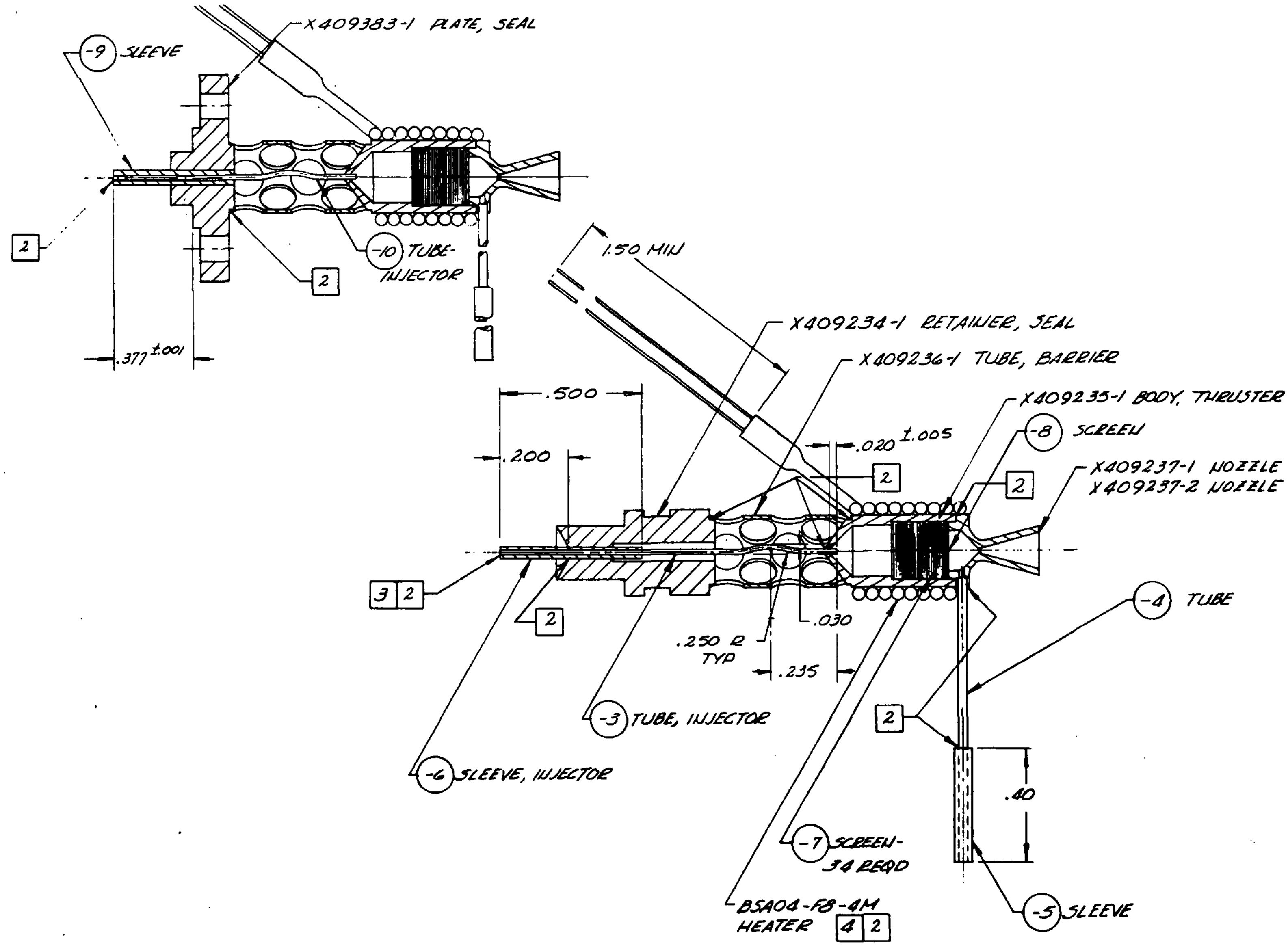
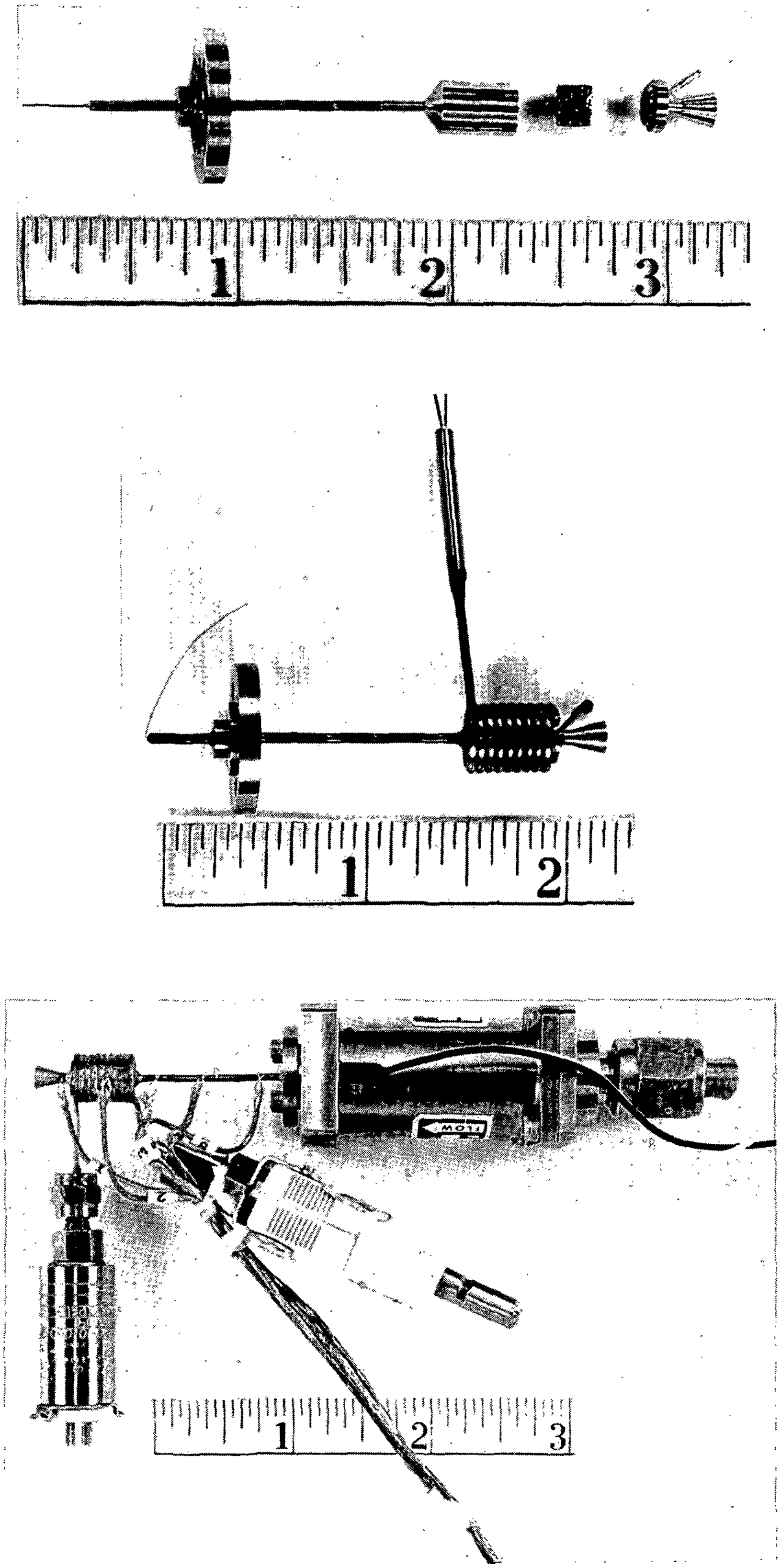


Figure 2-3. Thruster Internal Configuration



a) Components

b) Subassembly

c) Completed Unit

Figure 2-4. Development Model for Low Thrust Engine

- When the thruster valve is actuated, liquid hydrazine is admitted to the injector tube. Although a small amount of propellant vaporization may initially occur, the injector tube is designed to be rapidly quenched thermally.

- Liquid hydrazine enters the thrust chamber at relatively high velocity and contacts the heated screen pack (or flame holder).

- The hydrazine is heated to its vaporization point by contact with the screen pack and the hot gases circulating within the chamber.

- The temperature of the hydrazine vapor rises to the decomposition temperature. The decomposition occurs in accordance with the following reaction:

$$2N_2H_4 \longrightarrow 2NH_3 + N_2H_2$$

- A portion of the ammonia produced by the above reaction is endothermically dissociated into nitrogen and hydrogen by the following reaction:

$$2NH_3 \longrightarrow N_2 + 3H_2$$

- The decomposition products are exhausted from the reaction chamber through a supersonic propulsive nozzle.

Injector Design

The design of the injector is critical to the pulse reproducibility, the magnitude of chamber pressure oscillations during steady state operation, and the delivered specific impulse of the thruster. For a low-thrust, thermal-decomposition thruster designed for both steady state and pulsed operation, an optimum propellant injector should:

- Divide the propellant into small droplets that can be rapidly heated and vaporized and also distribute the propellant throughout the reaction chamber.

- Provide the necessary pressure drop to limit the instantaneous flow rate immediately after valve opening and also to decouple the supply system from chamber pressure oscillations.

- Provide a small dribble volume.
- Provide adequate resistance to contamination and material degradation for the design life of the thruster.

At the 20-millipound thrust design point, the pressure drop within the injector is 55 psi for an inlet pressure of 100 psia. A relatively large pressure drop is important for several reasons. At the instant of valve opening, the thrust chamber is evacuated. Because of the lack of chamber pressure, the instantaneous flow rate immediately after the valve is opened is much higher than the steady state flow rate. For injectors designed with very low pressure drops, the initial instantaneous flow rate may be several times greater than steady state. This can result in two undesirable effects. The chamber may become flooded or quenched with liquid propellant or else the excess propellant accumulated in the chamber during the start transient may decompose rapidly and result in a large pressure spike.

For steady state operation, the injector pressure drop is critical for a slightly different reason. If the injector pressure drop is very small, a small increase in chamber pressure can cause a large decrease in flow rate. This subsequently lowers the chamber pressure, which increases the flow rate. This phenomenon leads to the low-frequency "chugging" characteristic of some thrusters.

Because of the small flow rate, the injector of the electrothermal hydrazine thruster cannot be designed to produce a "shower-head" type of flow into the chamber. Instead, the incoming propellant stream is broken-up and, at least partially, vaporized by the hot decomposition gases circulating in the vicinity of the head end of the thruster.

Finally, the injector is designed to minimize the volume of liquid propellant between the valve seal and the thrust chamber. This is accomplished by the choice of length-to-diameter ratio and also by the method of attachment to the valve. Void volume reducers in the valve-to-injector seal allow the injector inlet to be located immediately downstream of the valve seat with essentially no excess volume. This is an important design criteria since the injector volume is emptied by "boiling" after valve closure. This phenomenon can greatly extend the thrust decay time because of the very low pressures and driving forces present as the dribble volume clears.

Thrust Chamber Design

The screen pack serves as the initial source of energy for propellant vaporization and decomposition. It also eliminates the line-of-sight flow path of liquid between the injector and the nozzle. A photograph of a screen pack for the hydrazine resistojet is shown in Figure 2-5. The unit is fabricated from Haynes 25, 40 x 40 mesh screen, welded together and ground to fit the inside diameter of the thrust chamber.

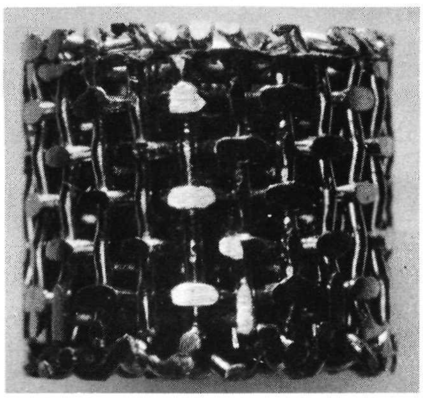


Figure 2-5. Screen Pack

Because the screen pack has a very low thermal capacity, the direct impingement of a continuous stream of propellant would rapidly quench the screen locally. This would result in channeling of liquid propellant through the screen pack, and erratic steady state performance. To prevent this effect, a large head space is provided in the vicinity of the propellant injector. A conical flow head space promotes recirculation of the hot decomposition gases. The recirculation is the result of a jet pumping effect of the propellant stream from the injector. The somewhat toroidal flow of decomposition gases around the propellant stream tube supplies some of the heat of vaporization to the propellant, and also helps distribute the propellant over the surface of the screen pack.

2.4 Thermal Design

The thermal design of the thruster utilized concepts from both electrothermal and monopropellant technology. Small physical size and low thermal mass, together with an effective insulation package, limit the heat losses so that performance is maximized and holding power is minimized. The heater geometry provides maximum reliability and is also compatible with the spacecraft bus voltage.

While in the quiescent state, the heat losses from the thruster are both conductive and radiative. Radiative losses are the result of insulation ineffectiveness and losses from the exposed inner contour of the propulsive nozzle. Insulation losses are minimized by making the thruster physically small and utilizing efficient insulation. Radiative losses from the nozzle can be reduced only by using low expansion area ratios. Since both stagnation temperature and nozzle area ratio affect the thruster performance, a compromise was necessary to determine the area ratio. Conductive heat losses arise from those portions of the thruster which attach to the heated chamber. The heat loss paths include the injector tube, the heater leads, thermistor or thermocouple leads and the mounting structure.

The thermal design of the injector is of critical importance to thruster operation. The injector is an obvious heat loss path and therefore the thermal conductance of the tube should be minimized. Thin wall tubing is used to produce the very large temperature gradient required by the low temperature interface at the valve and the high temperature interface at the thruster. The use of thin walled tubing also reduces the thermal mass of the injector and allows the incoming propellant to thermally quench the tube. This eliminates vaporization and/or decomposition in the tube. If vaporization occurs in the tube, the resultant increase in injector pressure drop suppresses the chamber pressure and degrades the thrust. This effect is not reproducible and leads to erratic performance.

As indicated previously, the thruster insulation package is molded from WRP-X-AQ. This material is supplied to TRW in the form of wet sheets which are subsequently molded around a mandrel, cured, and machined to shape. Figure 2-6 is a photograph of a completed insulation package.

Since the holding temperature of the thruster was established at a value less than the steady state operating temperature, a portion of the propellant decompositon energy is used to increase the temperature of the thruster. Since decomposition energy invested in sensible heat of the thruster, rather than the propellant, represents a performance loss, the thermal mass of the thruster was minimized. A small thermal mass also decreases the time required for the thruster to reach the holding temperature after the heater is activiated. The heat-up transient for a development unit is shown in Figure 2-7. The thruster reaches the minimum operating temperature of 800°F approximately 5 mintues after initiation of heater power.

2.5 Mechanical Design

The general configuration of the thruster was determined by performance and thermal criteria. The mechanical design effort concentrated on assuring that individual components and the completed assembly could survive the thermal and mechanical stresses the unit is expected to experience.

As indicated by Figure 2-5, the screen pack assembly is a rugged, all-welded unit fabricated from mesh material. Two techniques have been used in assembling the screen pack. Initially, a thin strip of Haynes 25 mesh material was rolled into a cylindrical shape and discs were welded on either end. Later units have used a stack of platinum gauze discs with a Haynes disc at the nozzle end of the assembly. Both methods have proven adequate from a structural standpoint; the platinum screen pack yields better performance.

As indicated by the callouts on Figure 2-3, the entire assembly is brazed together. Nicrobraze 210 powder is applied to the assembled components with suitable binder and stop-off materials. The unit is vacuum-brazed at 2150°F for 2 minutes. The braze remelt temperature is about 2100°F.

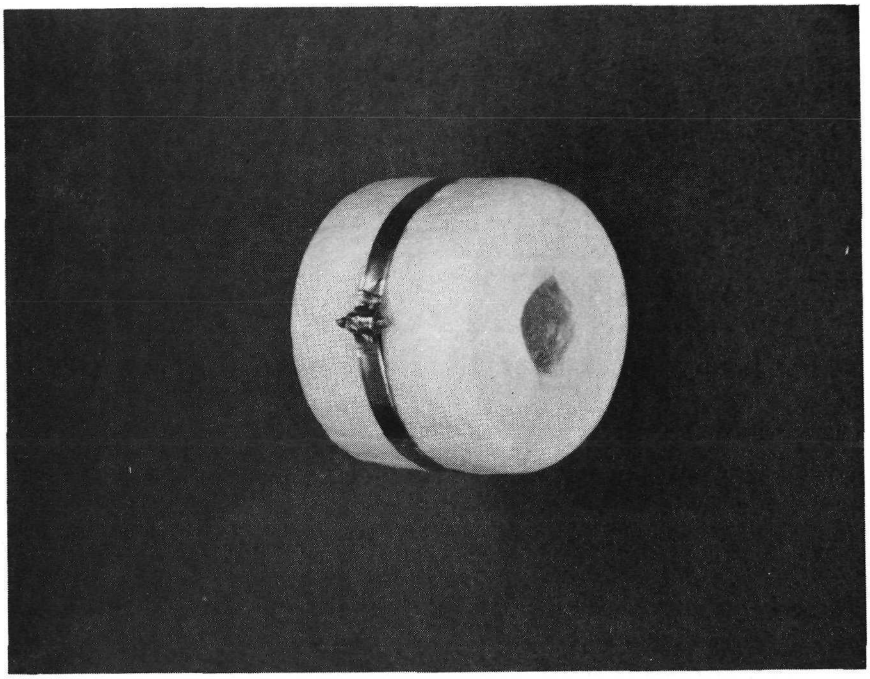


Figure 2-6. Insulation Package

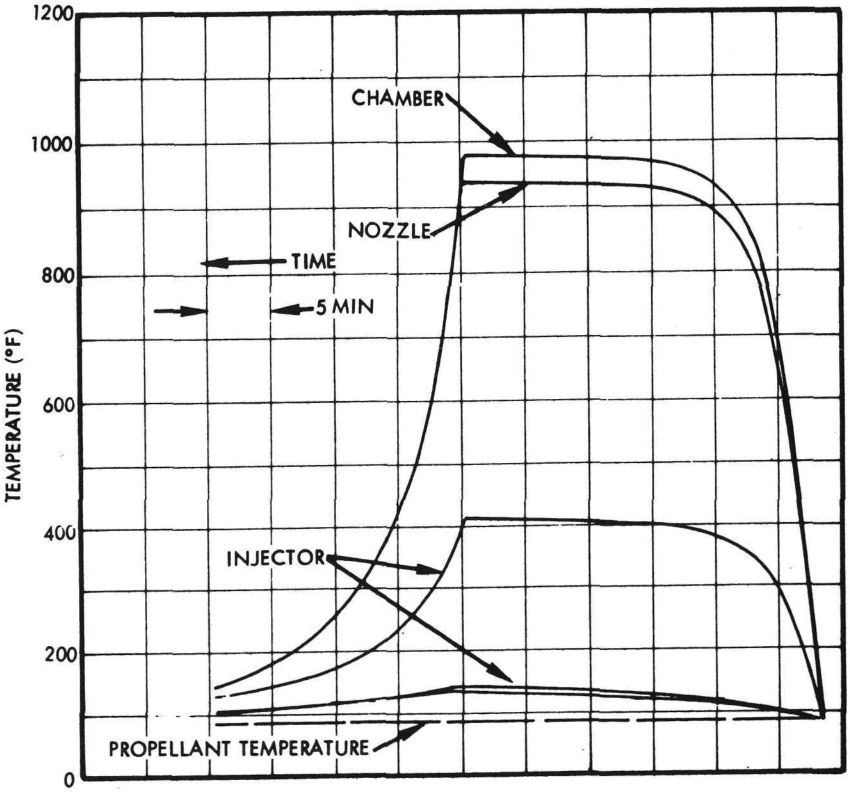


Figure 2-7. Thermal Response

The barrier tube is obviously designed with thermal considerations in mind. The hole pattern minimizes the heat losses through the tube yet allows the tube to retain sufficient strength to survive environmental loads.

The injector tube is designed with a "kink" to accommodate the stresses caused by thermal expansion of the barrier tube.

The insulation package is split longitudinally to allow removal from the molding mandrel and installation on the thruster. The method of attachment of the insulation is shown in Figure 2-6. A nickel ribbon is wound around the package and drawn tight by a key similar to that used to open a coffee can. The ribbon is spot welded and the key is cut off. The end of the key remains with the package as shown in Figure 2-6.

2.6 Electrical Design

The heater element geometry was determined by consideration of several factors. First, the unit selected had to be compatible with the thrust chamber external geometry. This more or less dictated the selection of a long tubular unit. Second, the specific configuration was selected on the basis of an excellent flight history on several TRW programs and the availability of a large amount of life test data. Finally, the tubular geometry is commercially available in a wide range of physical sizes and resistance values.

Two parallel resistance wires insulated with magnesium oxide are contained within an Inconel sheath. Nichrome V resistance wire was selected for this program although other materials with differing temperature/resistance characteristics are available. The junction between the large diameter lead wires and the small diameter resistance section is eliminated by using a differential swaging process during the manufacture of the unit.

Operation of the thruster at 28 Vdc and a power level of 5 watts results in a heater resistance requirement of 157 ohms. A tubular heater with an outside diameter of 0.025-inch and a length of 19 inches can satisfy this requirement. The heater will fit easily on the chamber (in two layers) with no increase in diameter relative to that shown in Figures 2-1 and 2-2.

The electrical connector for the heater has not been defined for lack of an electrical interface definition. It is anticipated that pigtail-type leads will be employed with a pin-and-socket splice such as that shown in Figure 2-8. The heater leads can be provided with a silicone rubber strain relief to preclude the possibility of lead breakage.

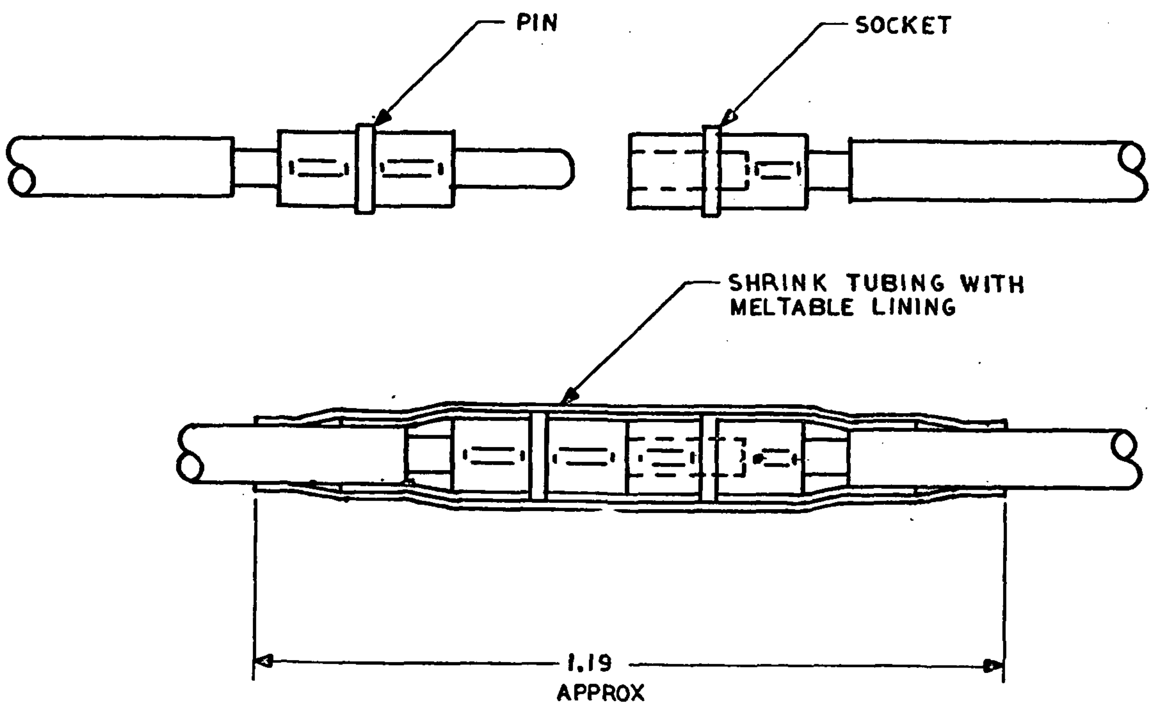


Figure 2-8. Pin and Socket Wire Splice

3. MATERIALS

Early in the program, the decision was made to make maximum use of materials already proven compatible with hydrazine. Based on experience with catalytic hydrazine thrusters, the most suitable material is L-605 (also called Haynes 25). Accordingly, all of the thruster components exposed to the propellant are fabricated from this alloy with one exception. Tbe screen pack is fabricated from platinum mesh.

Some of the early development thrusters utilized stainless steel injector tubes because of the lack of availability of small diameter Haynes tubing. Stainless steel was found to nitride severely in a relatively short period of time. Platinum-injector tubing was also used as an expedient since a wide range of sizes was available from another TRW program. Platinum proved structurally adequate for this purpose and no evidence of chemical incompatibility was observed. However, the use of platinum in this region of the thruster is contraindicated by its catalytic activity.

The experience gained with the use of platinum injectors, however, indicated that the material should prove useful for fabrication of screen packs. Tests with platinum screen packs yielded excellent performance.

No incompatibility problems have been observed with the braze alloy and other thruster materials.

Some doubt existed about the compatibility of the thruster insulation and the Inconel heater sheath. To resolve this question, a test was conducted in which WRP-X-AQ was placed in contact with Inconel 600 and maintained at 1700°F for 2 weeks. No significant interactions were observed.

4. PERFORMANCE

4.1 Pulsed Operation

The minimum impulse bit (MIB) is dependent upon a combination of minimum valve actuation time, the system supply pressure and response, the process time associated with the decomposition reaction, and several of the thruster design constraints. Figure 4-1 defines the various terms commonly used to describe thruster pulse characteristics. Another important parameter required to properly interface the propulsion system with a spacecraft control system is the time difference between the centroid of the valve command pulse and the centroid of the delivered impulse bit.

Ignition Characteristics

With the current thruster design, it is entirely feasible to obtain rise times of 15 milliseconds or less from valve signal to 90 percent of thrust. The rise times, of course, are not only a function of thruster design, but of valve response also. The holding temperature of the thruster is an important factor in determining the impulse bit and the specific impulse produced during a given pulse width.

Figure 4-2 shows the typical pulse shape obtained from a development thruster, S/N HRJ-9. (The photo is actually an overlay of ten consecutive pulses obtained after the thruster had achieved a quasi-equilibrium operation temperature.) The operating conditions were: inlet pressure, 105 psia; thrust level, 22.7 millipounds; command pulse width, 50 milliseconds; valve open time, 60 milliseconds. As indicated by the figure, the rise time to 90 percent of steady state chamber pressure is about 12 milliseconds. The upper trace is the valve current, which is measured by the voltage across a calibrated series resistor. While the opening time is clearly indicated, the closing of the valve is not visible in this photo. Valve closure is very nearly simultaneous with the start of chamber pressure decay.

Although the thrust rise times are very consistent, the total impulse produced per pulse is a function of thruster temperature. This variation in delivered impulse per pulse with temperature can be minimized

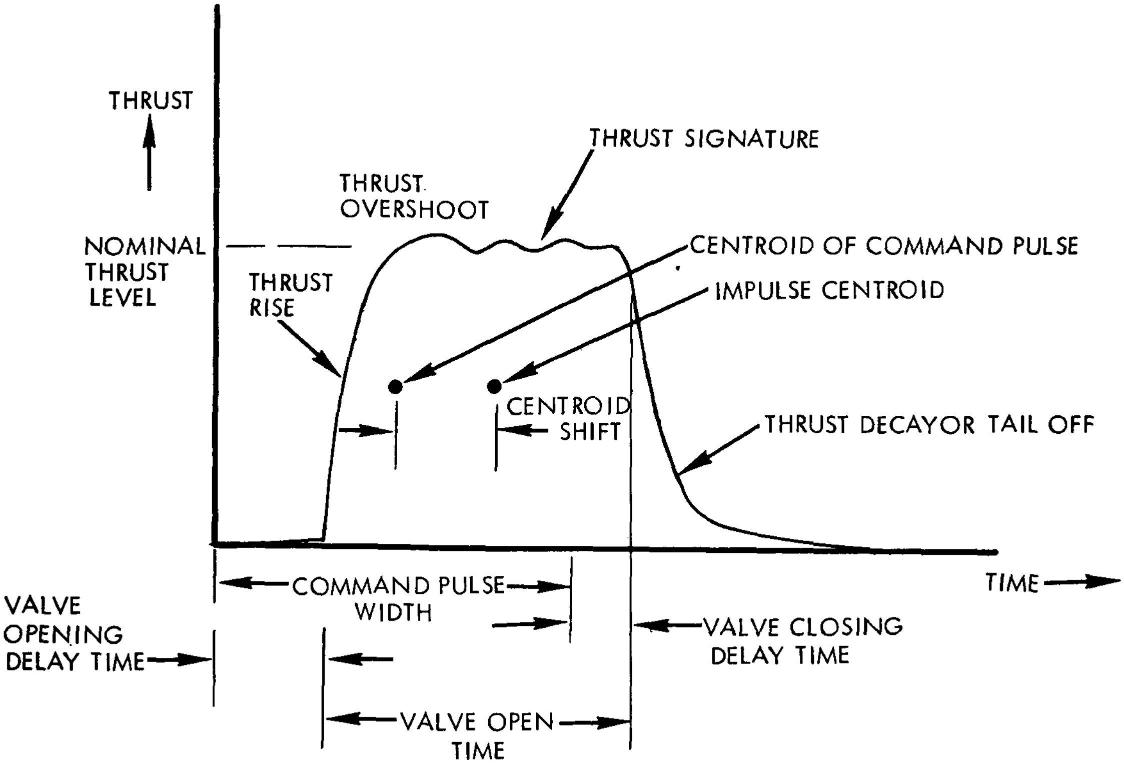


Figure 4-1. Definition of Pulse Parameters

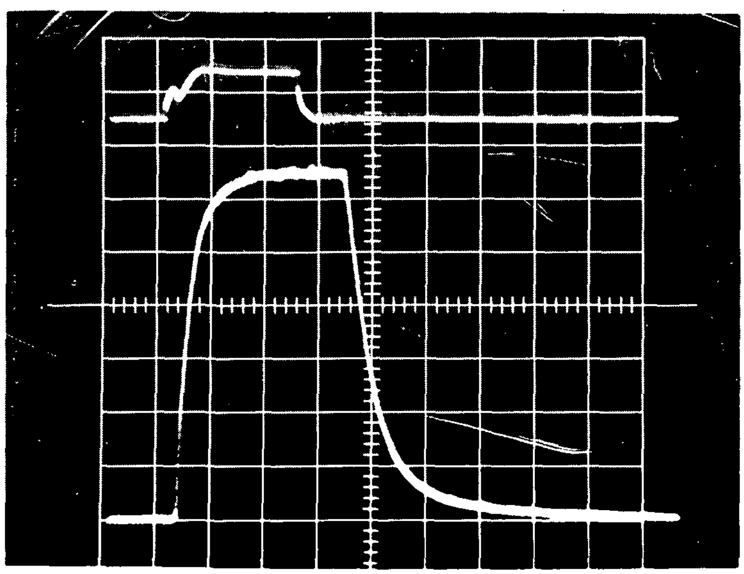


Figure 4-2. Chamber Pressure Trace of 0.050 Second Pulse

by increasing the holding temperature so that it more closely coincides with the adiabatic flame temperature. The higher holding temperatures minimizes the "heat-sink" effect of the screen pack and thrust chamber. In simplified terms, this energy loss may be represented by:

$$q = WC\ (T_F - T_0)$$

where

- q = Energy lost to heat the thruster, BTU
- W = Weight of heated parts, lb
- C = Average heat capacity of heated parts
- T_0 = Holding Temperature, °R
- T_F = Final Temperature, °R

Thus, for a thruster which operates predominately in the pulsed mode, the performance can be maximized by minimizing the thruster weight, limiting the number of starts from the holding temperature, or else increasing the holding temperature.

The auto-decomposition temperature of hydrazine in the pressure ranges of interest is about 500°F. A large number of tests has shown that ignition consistently occurs within small thermal decomposition thrusters, in which the residence times are relatively short, at temperatures of about 800°F.

Thrust Decay Time

The time for the chamber pressure (and thrust) to decay from the steady state value to zero pressure is influenced by a large number of factors, including the injector design and dribble volume, the thrust chamber shape and characteristic length, the chamber temperature and the chamber pressure. When the thruster valve is closed, the propellant remaining in the injector and thrust chamber is expelled somewhat inefficiently, since the driving pressure of the feed system is no longer effective. Examination of Figure 4-2 indicates that the expulsion of this propellant is accomplished at measurable pressures for more than 100 milliseconds, even though decay to 10 percent P_c occurs within about 20 milliseconds.

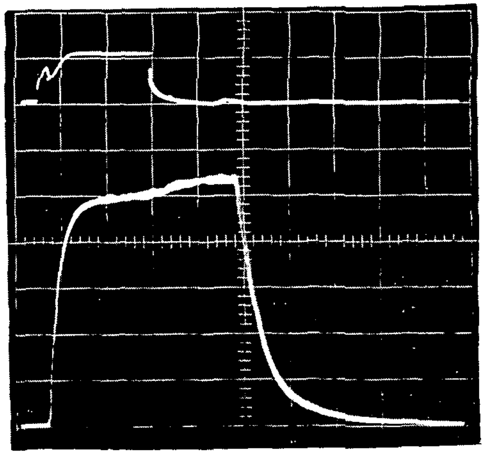
Pulse Reproducibility

The repeatability of pulse shape and delivered impulse bit from pulse to pulse is very important for applications in which multiple thrusters are used for spacecraft control. The EHT is ideally suited to such applications because of the precise pulse characteristics it produces. The pulse reproducibility of thruster S/N HRJ-7 is shown in Figure 4-3. The photos on the left are overlays of the chamber pressure traces of 10 consecutive hot pulses. The photos on the right are overlays of pulse sequences using different command pulse widths. For this test sequence, thruster inlet pressure was the primary variable. An inlet pressure of 210 psia produced a thrust level of 55 millipounds. The inlet pressure was reduced incrementally to yield progressively lower thrust levels. The final two photos were taken with the inlet pressure set at 27 psia. The resultant thrust level was 5 millipounds. This test sequence illustrates not only the excellent reproducibility of consecutive pulses under near constant conditions, but also shows the effects that can be attributed to the propellant supply system blowdown pressure range. The pulse transients and centroids are nearly invariable with supply pressure, but the magnitude of the impulse bit varies nearly linearly with inlet pressure.

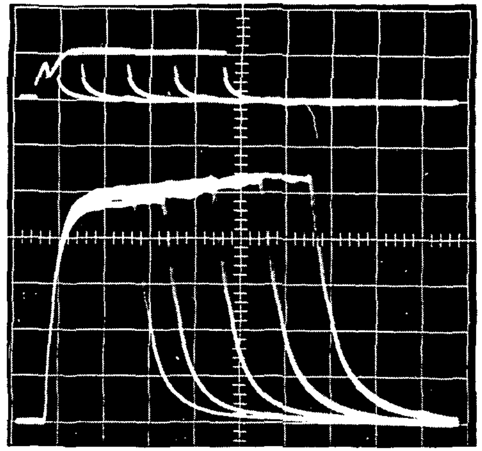
4.2 Pulsed Performance

For many applications, a large portion of the thruster total impulse will be delivered in a pulsed mode of operation. Further, much of this total impulse may be delivered in a large number of very small, discrete impulse bits with long off times. For monopropellant hydrazine thrusters in general, this is an inherently inefficient mode of operation.

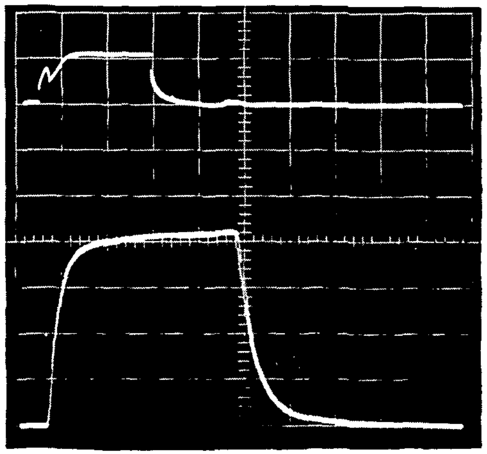
Figure 4-4 is a reproduction of an oscillograph trace of typical pulsed test data taken with the thruster shown in Figure 2-4. The boundary conditions are: injection pressure, 250 psia; holding temperature, 1200°F; electrical pulse width, 0.050 second.



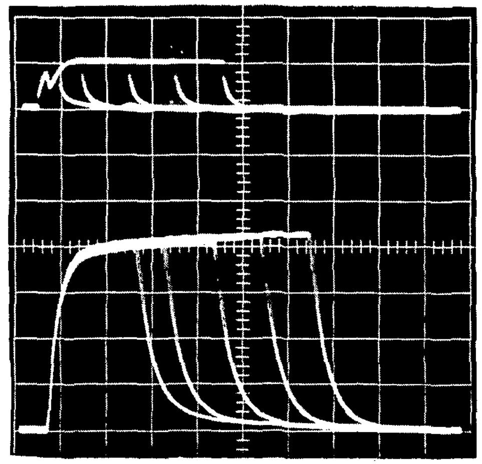
10 PULSES 50 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE



3 - 5 PULSES EACH 10 - 20 - 40 - 60 - 80 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE



10 PULSES 50 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE



3 - 5 PULSES EACH 10 - 20 - 40 - 60 - 80 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE

Figure 4-3. Pulse Characteristics of HRJ-7

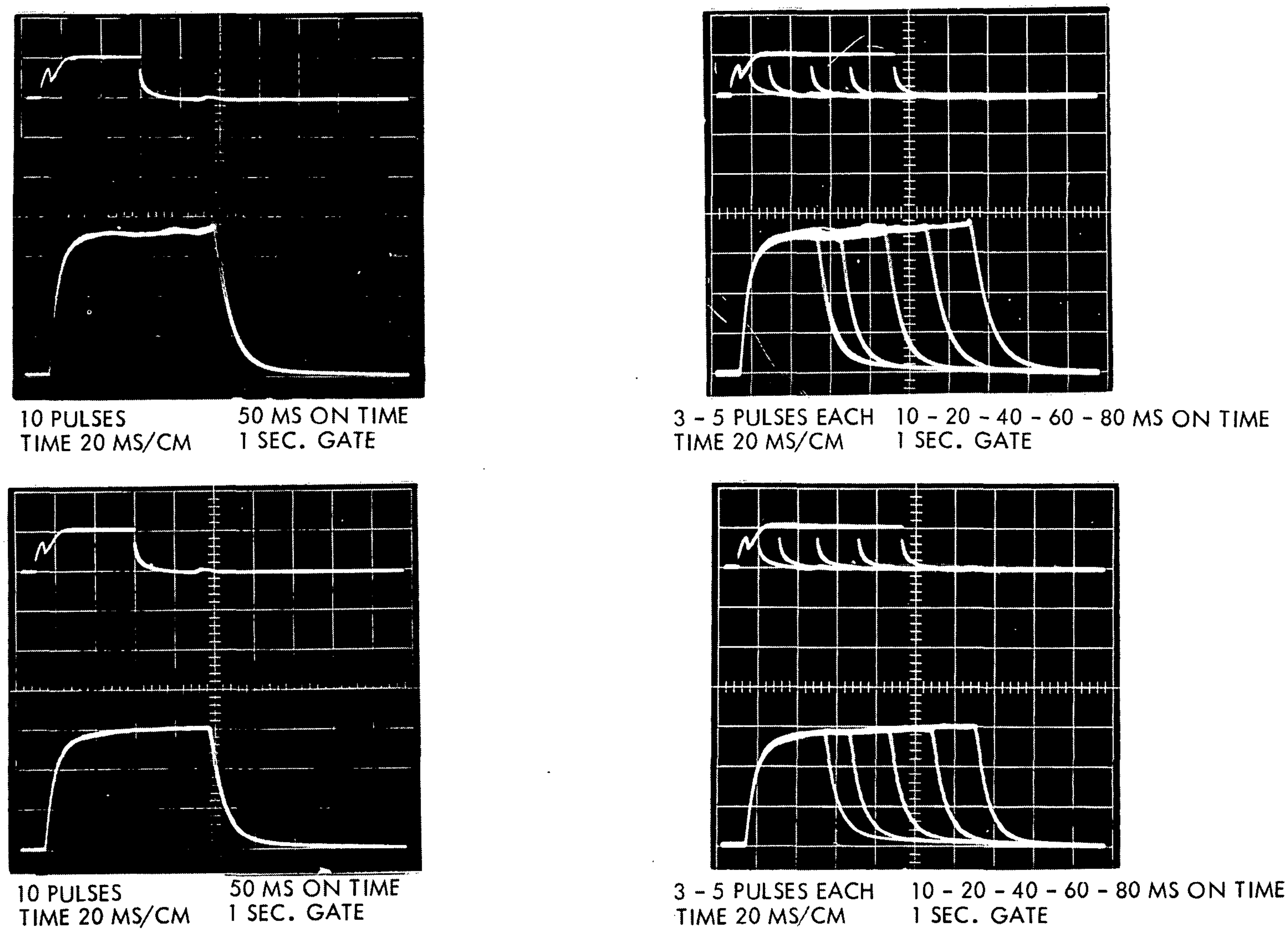
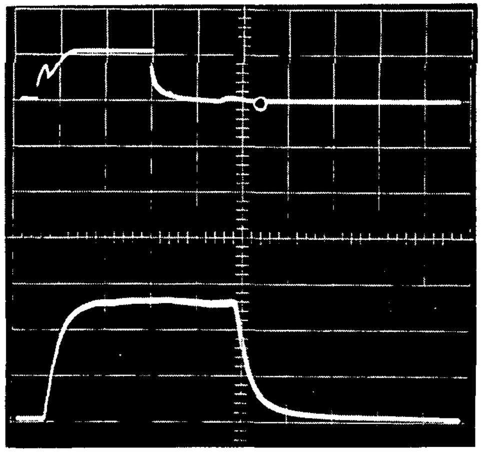
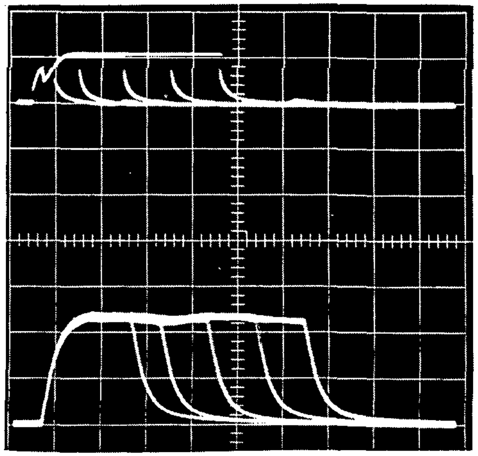


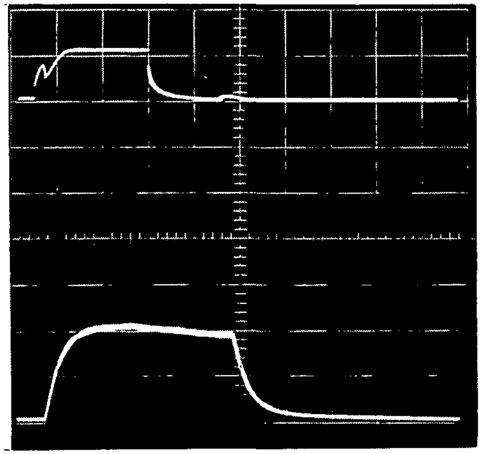
Figure 4-3. Pulse Characteristics of HRJ-7 (Continued)



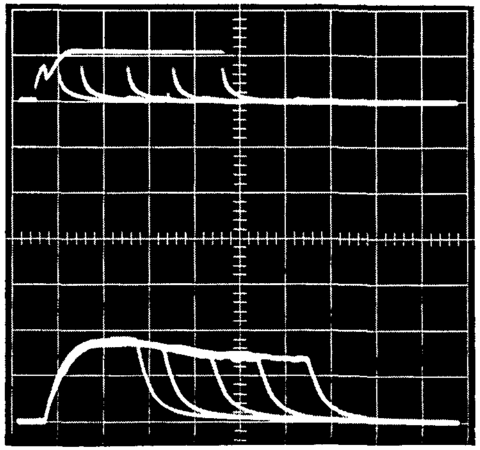
10 PULSES 50 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE



3 - 5 PULSES EACH 10 - 20 - 40 - 60 - 80 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE

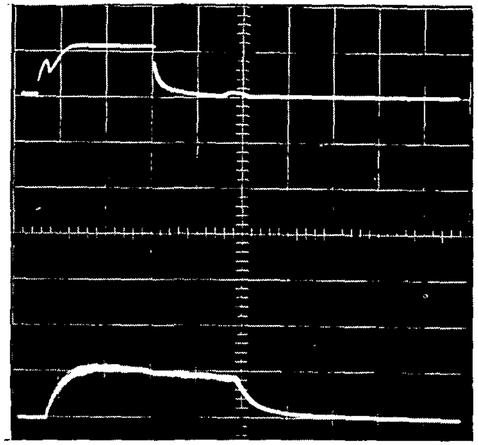


10 PULSES 50 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE

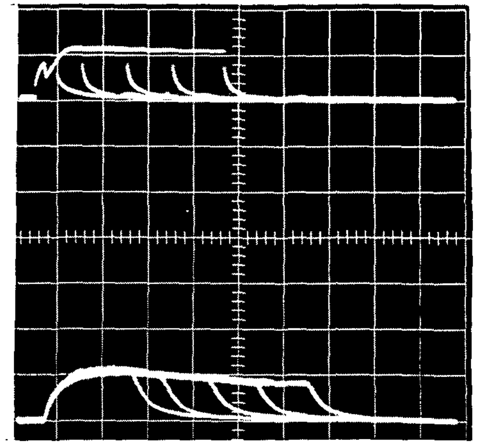


3 - 5 PULSES EACH 10 - 20 - 40 - 60 - 80 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE

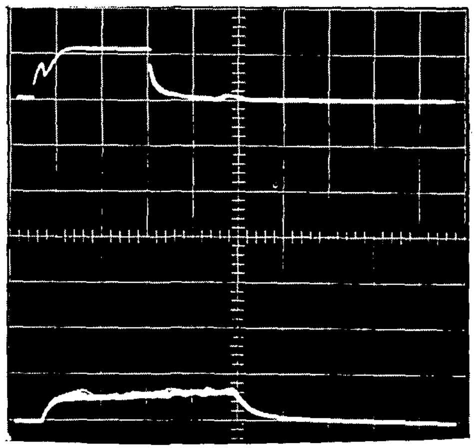
Figure 4-3. Pulse Characteristics of HRJ-7 (Continued)



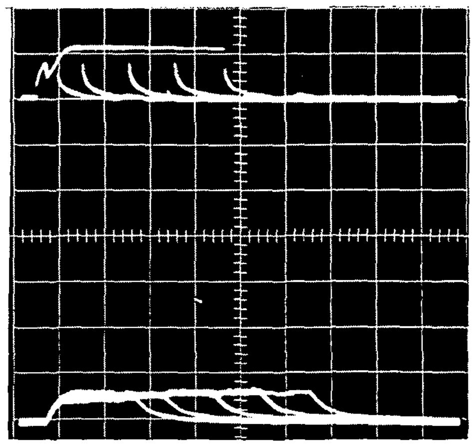
10 PULSES 50 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE



3 - 5 PULSES EACH 10 - 20 - 40 - 60 - 80 MS ON TIME
TIME 20 MS/CM 1 SEC . GATE



10 PULSES 50 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE



3 - 5 PULSES EACH 10 - 20 - 40 -60 - 80 MS ON TIME
TIME 20 MS/CM 1 SEC. GATE

Figure 4-3. Pulse Characteristics of HRJ-7 (Continued)

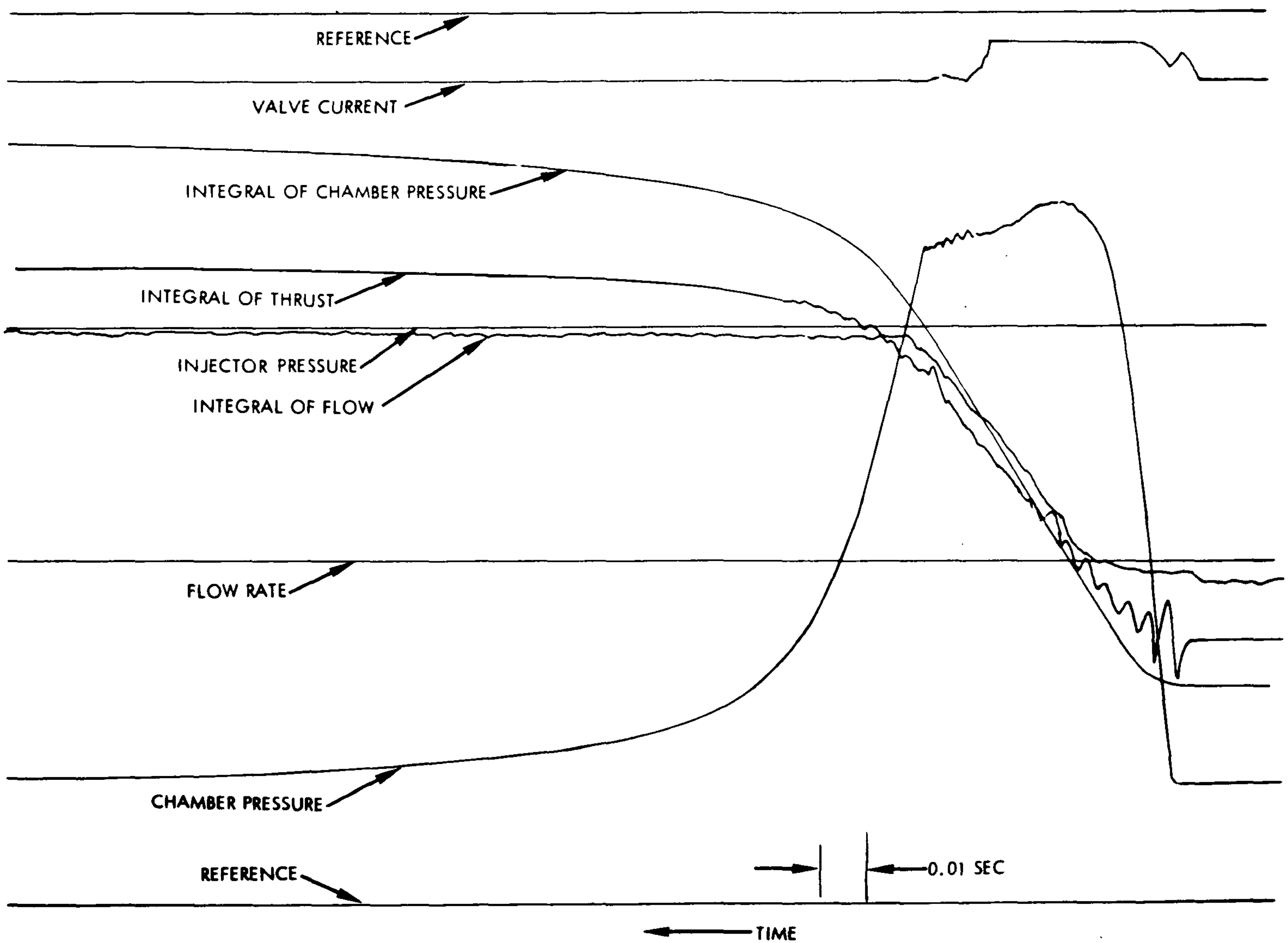


Figure 4-4. Typical Pulse Data

The primary nomenclature on the trace is:

I_v,	Valve current, which also indicates the opening and closing times of the valve and the electrical pulse width.
P_{inj},	The injection pressure measured at the inlet to the valve.
P_c	The thrust chamber pressure.
$\int P_c$	The integral of the chamber pressure.
$\int F$,	The integral of thrust, more commonly called impulse bit.
$\int$flow,	The amount of propellant expended during the pulse.
Flowrate,	Shows the total travel of the flow measurement device.

All of the critical pulse mode parameters can be obtained from the data contained on the oscillograph chart, together with the slowly varying temperature data recorded on strip chart recorders. The reduced performance parameters from this single oscillograph record are listed in Table 4-1.

Table 4-1. Pulse Data

Integral of Chamber Pressure	8.730 psia-seconds
Integral of Thrust	3.227×10^{-3} lb-sec
Integral of Flow	1.957×10^{-5} lb(m)
Specific Impulse	164.9 sec
C*	3452 ft/sec
C_F	1.536
Inlet Pressure	250 psia
Nominal Steady State Chamber Pressure	125 psia
Nominal Steady State Thrust	5.0×10^{-2} lb
Pulse Width	0.05 sec
Pulse Gate	2.0 sec
Rise Time (to 90 percent)	0.012 sec
Decay Time (to 10 percent)	0.027 sec

Figure 4-5 indicates the worst case, low-duty-cycle performance of the thruster shown in Figure 2-4. The holding temperature was 1200°F. The electrical pulse width was 0.050 second and the off time was 1.950 seconds. The delivered impulse bit varied from 0.00306 lb-sec to 0.00314 lb-sec. This variation is within the measurement accuracy of the data acquisition system. The nominal specific impulse for this test was 161 seconds. Higher duty cycles improve the performance.

4.3 Cycle Life Capability

As stated earlier, a cycle life test was conducted using development hardware. Figure 4-6 shows some of the data taken during this test. Each figure is an overlay of 10 to 20 consecutive chamber pressure traces taken at different times during the test. For all 10 photos, most of the test parameters are identical. The electrical pulse width is 0.030 second and the OFF time is 0.470 second. The thruster inlet pressure is 100 psia. Because a wide range of operating conditions were incorporated in the test, the thruster operating temperature is not entirely consistent from test to test. This variability in temperature accounts for some of the change in pulse shape and response times. At 200 pulses, the thruster was relatively cool and produced rapid rise times to steady state. In all subsequent photos, the injector temperature was quite high and the thrust rise times were relatively long. This was a design deficiency for the unit tested and has since been corrected. In the final photo, taken at 1,001,400 cycles, the initial transient has a spike in it. This spike has been observed in other thrusters and is attributed to propellant vaporization in the injector and subsequent quenching. As mentioned before, this problem has been corrected by a thermal redesign of the propellant injector.

Several significant conclusions can be drawn from this test. A cycle life capability has been demonstrated which is equal to the contractural requirement. Based on the integrated area under the chamber pressure curves, the delivered impulse bit variation was less than $\pm$ 15 percent over the 10^6 cycles. Some of the variation is attributed to non-reproducibility of test conditions and measurement errors. The shift in the pulse centroid was very small, accounting to less than 5 milliseconds. Finally, no evidence of injector plugging was observed upon disassembly of

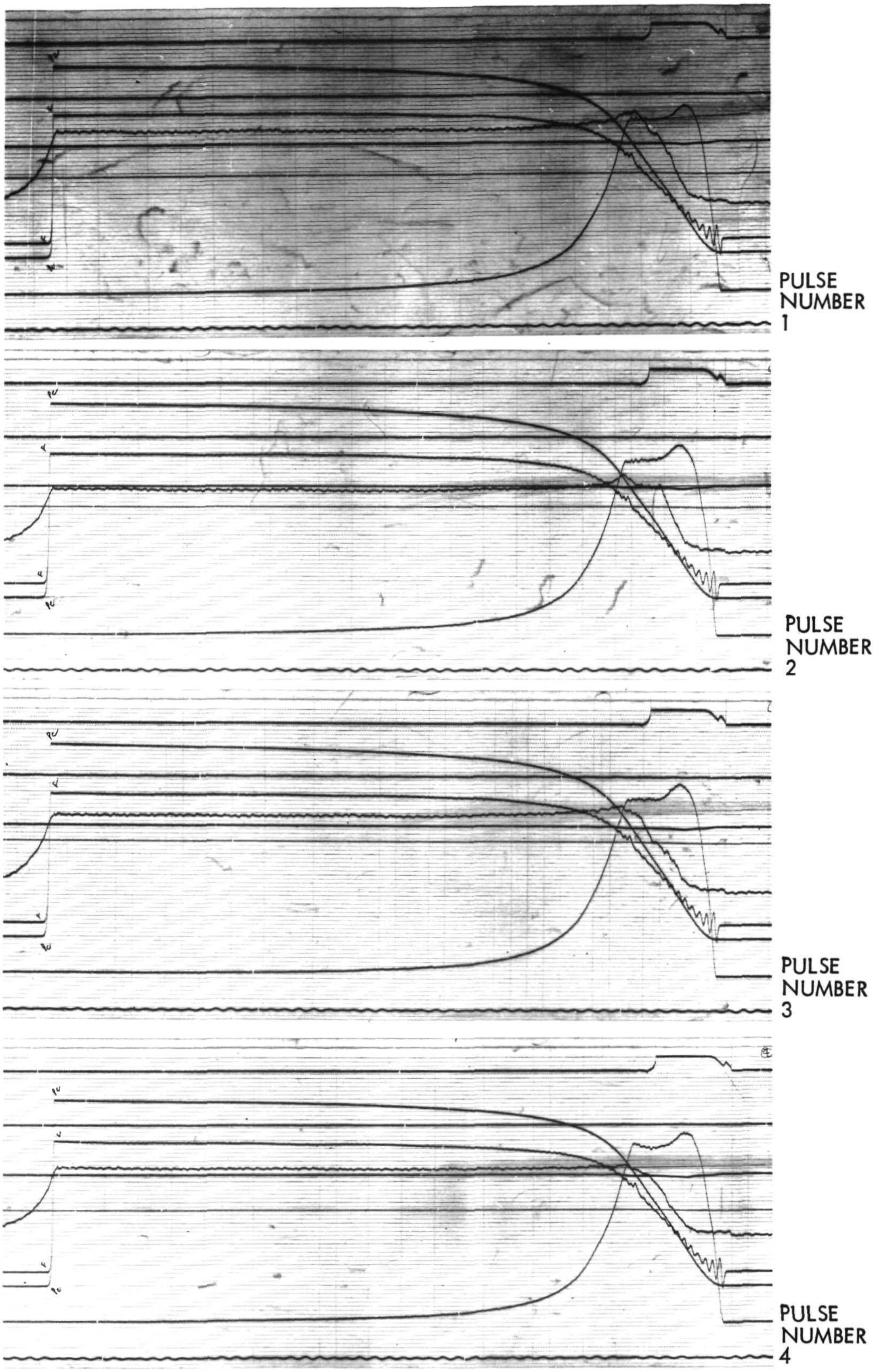
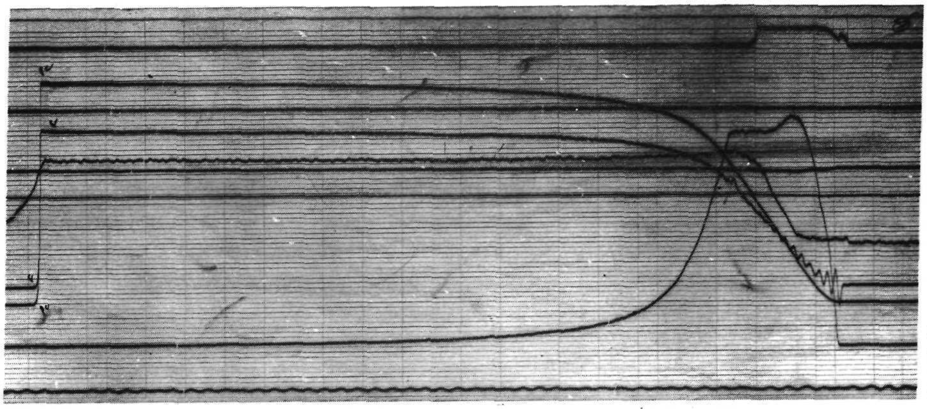
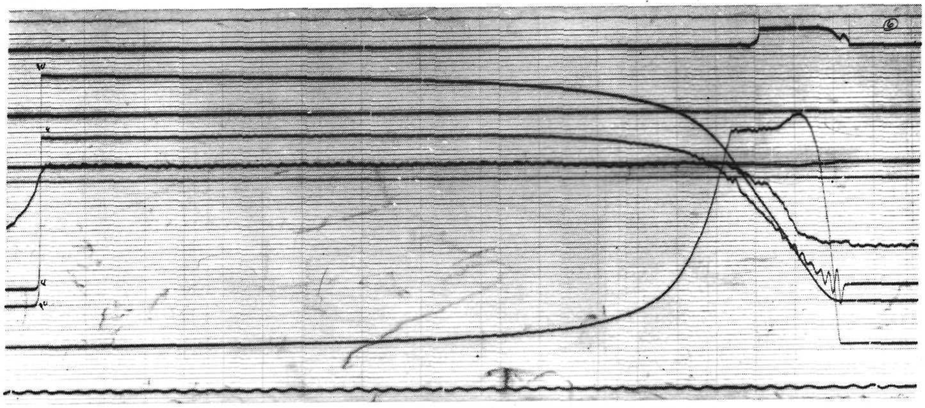


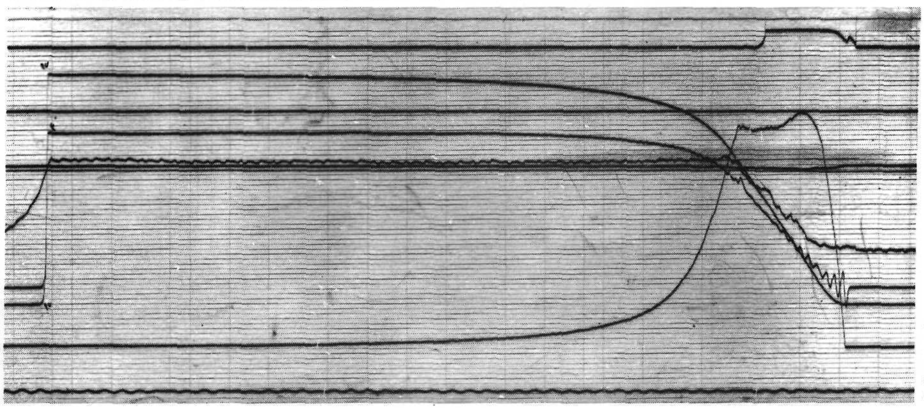
Figure 4-5. Pulse Data at Low Duty Cycles



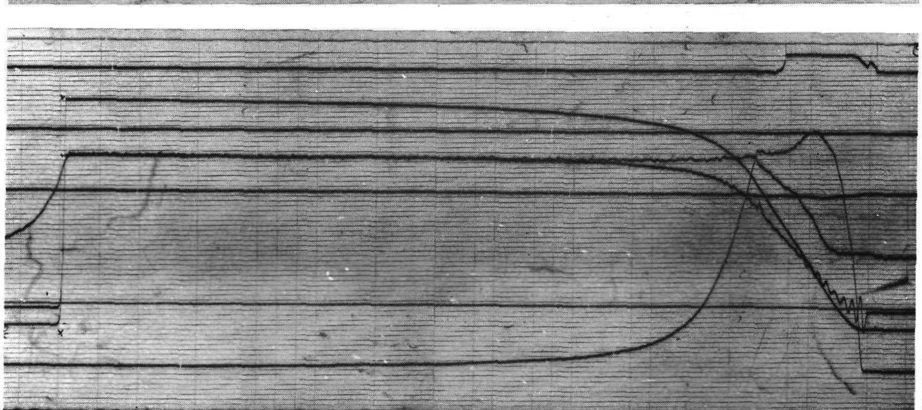
PULSE NUMBER 5



PULSE NUMBER 6

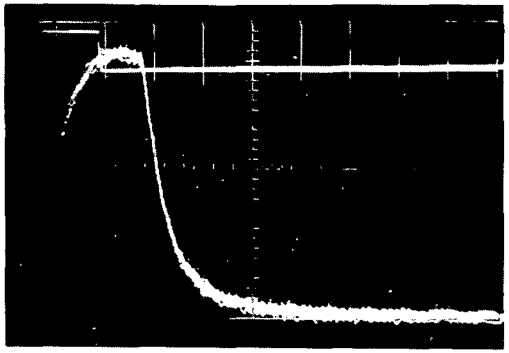


PULSE NUMBER 7

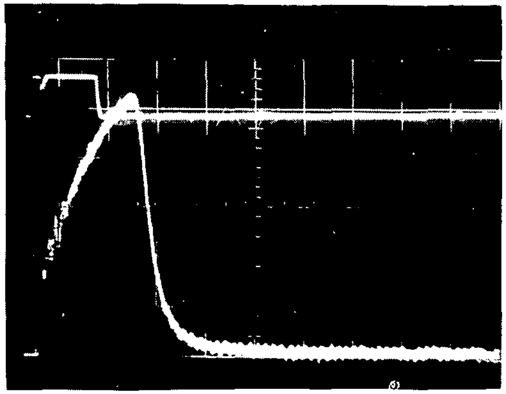


PULSE NUMBER 8

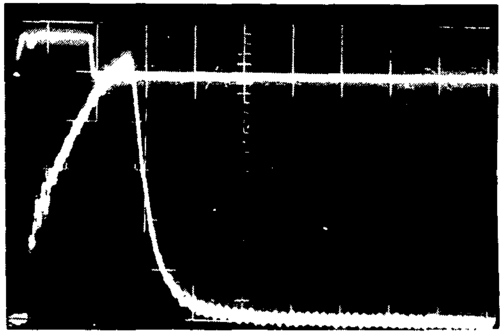
Figure 4-5. Pulse Data at Low Duty Cycles (Continued)



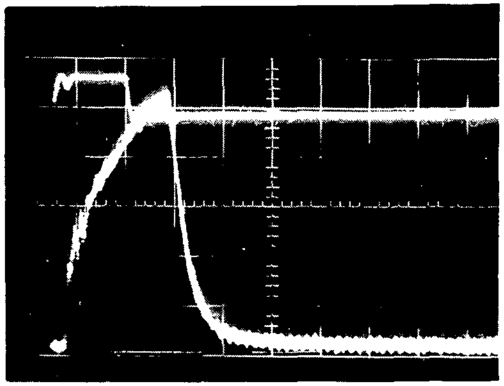
200 Pulses



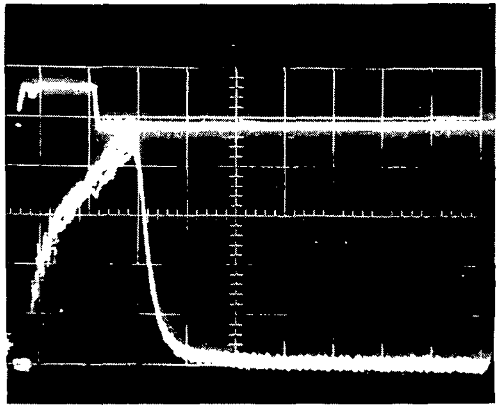
110,000
Pulses



150,500
Pulses

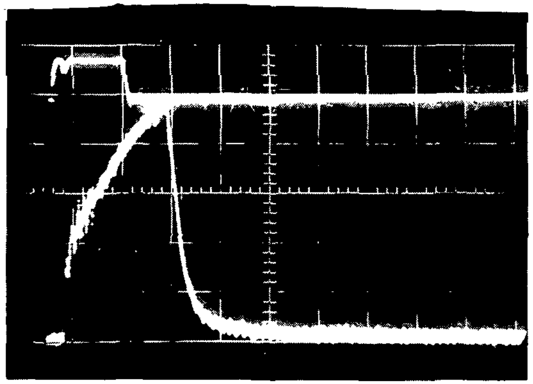


202,500
Pulses

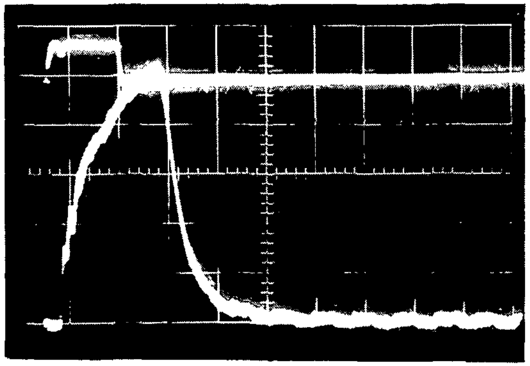


259,100
Pulses

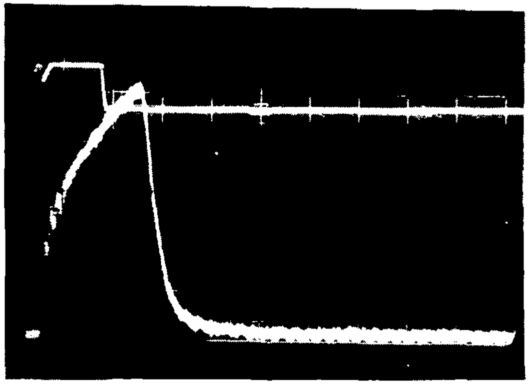
Figure 4-6. Cycle Life Data



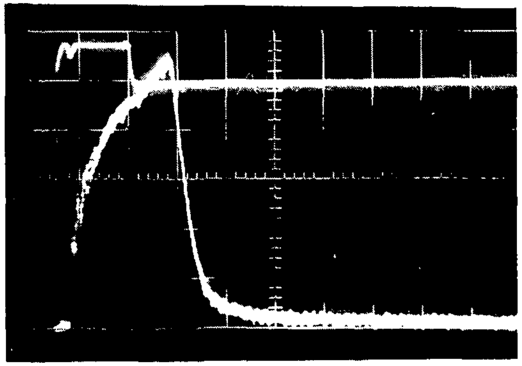
307,000
Pulses



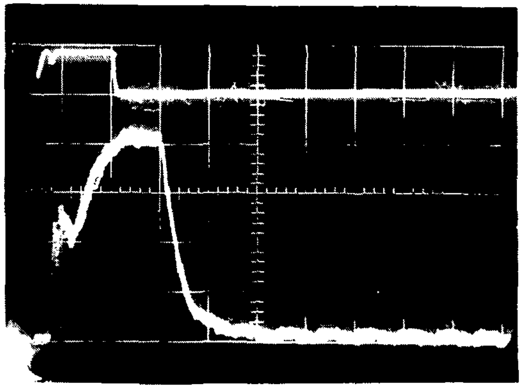
356,000
Pulses



52300 Pulses



701,000
Pulses



1,001,400
Pulses

Figure 4-6. Cycle Life Data (Continued)

the thruster. The test was conducted with mil-grade hydrazine. A propellant analysis is given in Table 4-2.

4.4 Steady State Operation

Since the pulsing requirement is by far the most stringent from a design point, little has been said relative to steady state operation. For the EHT, which must operate in both modes, care must be taken not to over-compromise one in favor of the other.

In addition to specific impulse, one criteria commonly used to judge the design adequacy of a steady state thruster is chamber pressure roughness. The roughness exhibited by Thruster S/N HRG-9 is shown in Figure 4-7. The nominal value for chamber pressure is 74 psia which results in a measured thrust level of 30 millipounds. The variation in pressure is nominally about $\pm$ 2 psia. These oscillations could be damped somewhat by enlarging the chamber volume. Thrusters with large values of L* are not compatible with good pulsing operation, however.

Since the holding temperature is characteristically less than the steady state operating temperature, the specific impulse approaches a steady-state maximum asymptotically. Characteristic transients in specific impulse are shown in Figure 4-8 for thruster series number HRJ-8. The holding temperature was 920°F. Figure 4-9 shows the steady state specific impulse delivered by the thruster shown in Figure 2-4. This thruster is more representative of the current design. The measured chamber temperature range from 1680°F at the lowest inlet pressures to 1740°F at the maximum inlet pressure.

4.5 Test Methods

All of the specialized test equipment required for the test program is currently operational. The test equipment was developed for use on TRW's Independent Research and Development electrothermal hydrazine thruster program.

The vacuum system has a chamber four feet in diameter and four feet long. The pumping system employs two 10-inch diffusion pumps and a single 80-cfm rotary mechanical pump. While this system has proven adequate to maintain the environmental pressure in the 10^{-5} range during pulse mode

Table 4-2. Propellant Analysis for Million Cycle Test

	% Weight	PPM
N_2H_4	99.21	
H_2O	0.33	
$CO_2 + NH_3$	0.33	
Unidentified (C_xF_x)	0.13	
Non-Volatile Residue		23.2
Cu		0.3
Fe		2.3
Zn		0.4
Mg		0.3
Al		3.0
Na		Present
K		2.5
Ca		1.6
Ni		0.6
Chloride		9.6

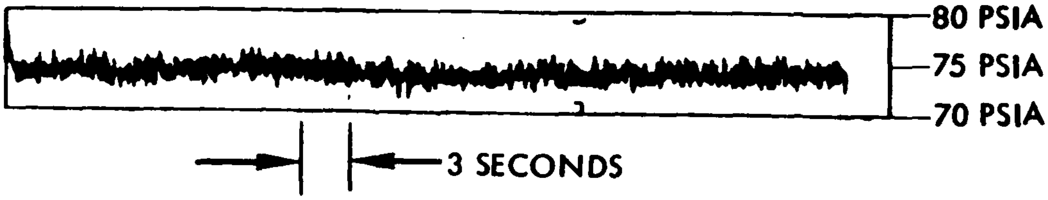


Figure 4-7. Chamber Pressure Roughness

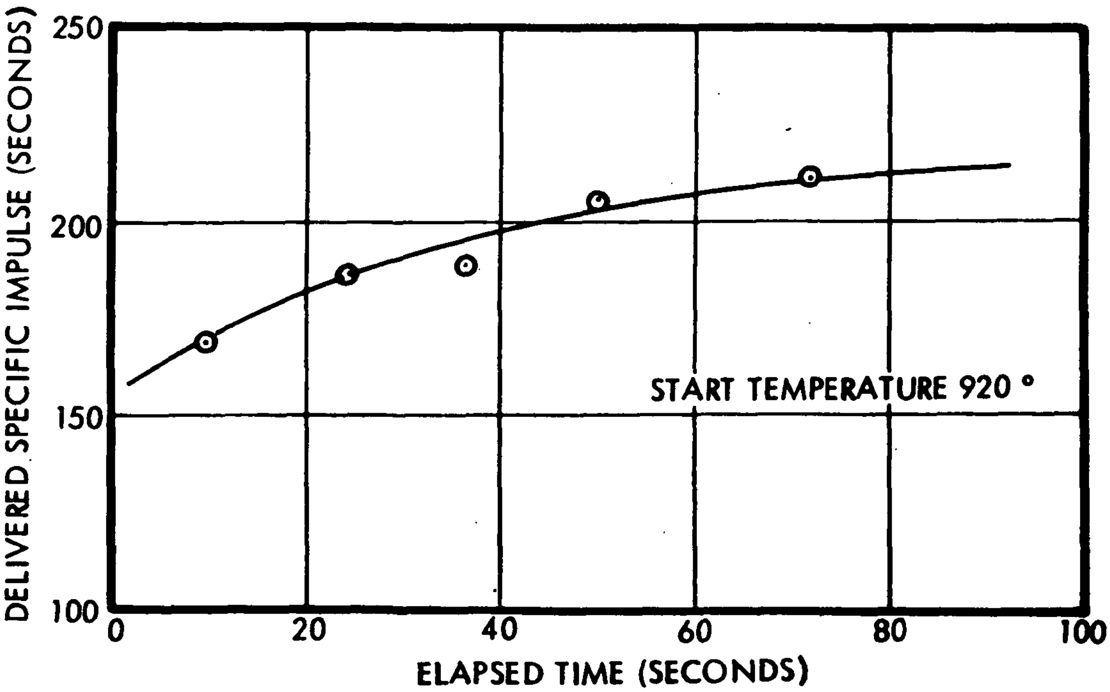


Figure 4-8. Specific Impulse Transient

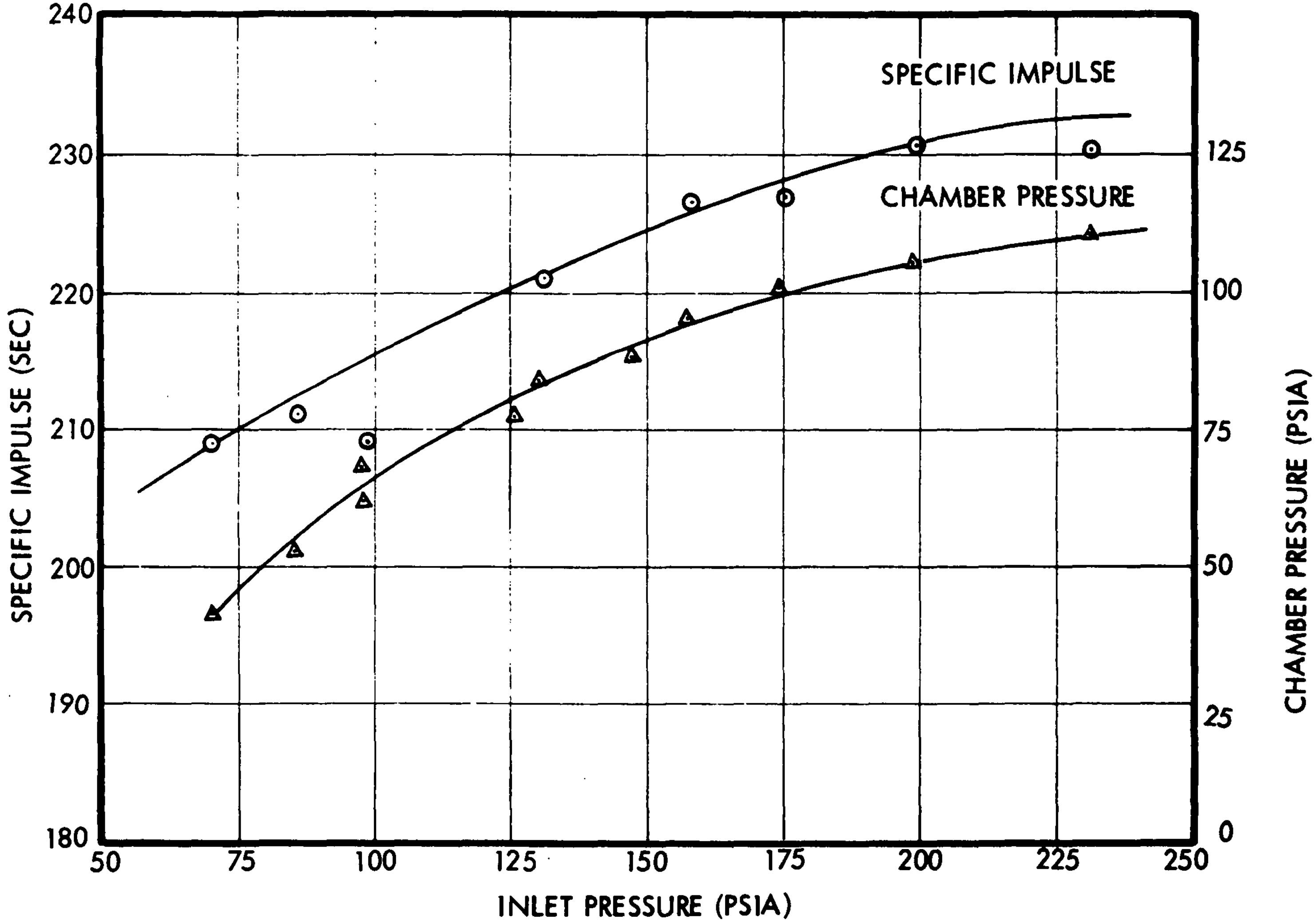


Figure 4-9. Steady State Performance

tests, steady state tests must be accomplished with the rotary pump alone. For this reason, all of the test equipment will be transferred to an existing facility with a pumping speed sufficient to maintain the environmental pressure below 10^{-4} torr under steady state conditions.

The thrust balance currently in use, shown in the lower left hand portion of Figure 4-10, has a natural frequency which is high enough to allow the measurement of the thrust produced by consecutive pulses for any duty cycle applicable to the EHT. Steady state thrust can also be measured. The balance is a parallelogram design, wherein the thruster mounting platform is supported by four flexures. Movement of the platform is limited to a few microinches by a load cell.

Flow measurements are accomplished with the apparatus shown in the right hand portion of Figure 4-10. The calibrated burette contains a Teflon sealed piston which is pressurized with nitrogen. As propellant is expelled, the position of the piston is monitored by a linear displacement potentiometer. Burettes of different capacities can be easily installed for various pulse widths or steady state operation. A system of solenoid valves allows thruster operation from either a main propellant supply or from the burette supply. With this capability, the burette can be refilled remotely without the need for turning off the thruster or interrupting the test.

A new data acquisition system has been assembled and checked out specifically for testing of electrothermal hydrazine thrusters. In effect, the facility is comprised of a series of integrating amplifiers which condition the signals from the chamber pressure transducer, the thrust balance, and the propellant flowmeter. Figure 4-11 is a photo of the data acquisition system. All of the specific impulse data presented for the electrothermal hydrazine engine is the result of direct measurement of thrust and propellant flow rate, for both pulsed and steady state conditions.

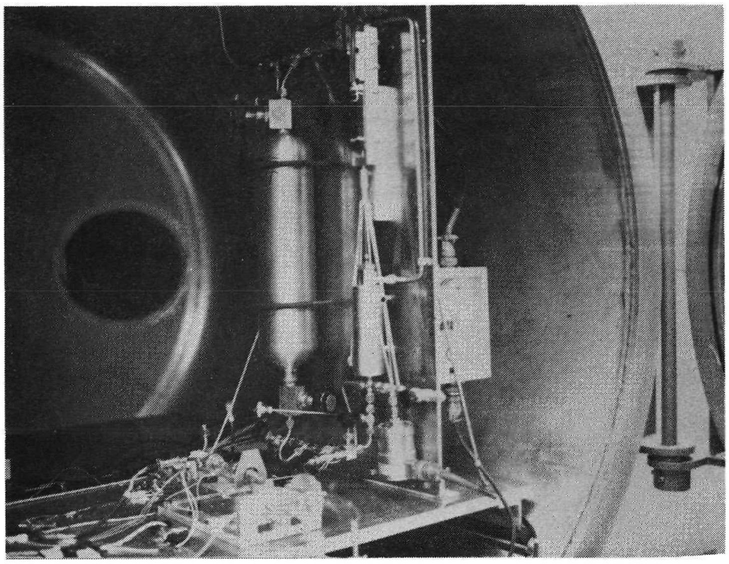


Figure 4-10. Test Apparatus

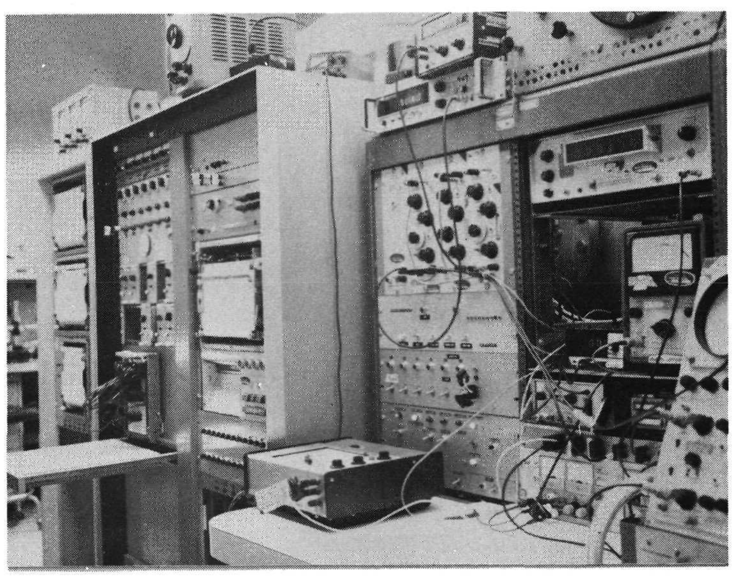


Figure 4-11. Data Acquisition System

5. CONCLUSIONS

Based on the design and performance data contained herein, the following conclusions can be drawn.

- The thruster design simple and inherently reliable.
- Either of the preliminary EHT designs (Ref. Figures 2-1 and 2-2) are capable of fulfilling the requirements of the Statement of Work.
- Compared to small catalytic thrusters, EHT offers significantly better performance in terms of delivered specific impulse, impulse bit size and reproducibility, and thruster longevity.
- With just two exceptions, all of the requirements that must be fulfilled during the Engineering Model test have been demonstrated with development hardware. The steady state life of 30 hours remains to be proven and operation at 27 Vdc has not yet been demonstrated.
- Based on data generated to date, the EHT is capable of rapid flight qualification.