NASA/CR—2001-210709

Mars Flyer Rocket Propulsion Risk Assessment

ARC Testing

Atlantic Research Corporation
Niagara Falls, New York

April 2001

The NASA STI Program Office . . . in Profile

Since its founding, NASA has been dedicated to the advancement of aeronautics and space science. The NASA Scientific and Technical Information (STI) Program Office plays a key part in helping NASA maintain this important role.

The NASA STI Program Office is operated by Langley Research Center, the Lead Center for NASA's scientific and technical information. The NASA STI Program Office provides access to the NASA STI Database, the largest collection of aeronautical and space science STI in the world. The Program Office is also NASA's institutional mechanism for disseminating the results of its research and development activities. These results are published by NASA in the NASA STI Report Series, which includes the following report types:

- TECHNICAL PUBLICATION. Reports of completed research or a major significant phase of research that present the results of NASA programs and include extensive data or theoretical analysis. Includes compilations of significant scientific and technical data and information deemed to be of continuing reference value. NASA's counterpart of peer-reviewed formal professional papers but has less stringent limitations on manuscript length and extent of graphic presentations.

- TECHNICAL MEMORANDUM. Scientific and technical findings that are preliminary or of specialized interest, e.g., quick release reports, working papers, and bibliographies that contain minimal annotation. Does not contain extensive analysis.

- CONTRACTOR REPORT. Scientific and technical findings by NASA-sponsored contractors and grantees.

- CONFERENCE PUBLICATION. Collected papers from scientific and technical conferences, symposia, seminars, or other meetings sponsored or cosponsored by NASA.

- SPECIAL PUBLICATION. Scientific, technical, or historical information from NASA programs, projects, and missions, often concerned with subjects having substantial public interest.

- TECHNICAL TRANSLATION. English-language translations of foreign scientific and technical material pertinent to NASA's mission.

Specialized services that complement the STI Program Office's diverse offerings include creating custom thesauri, building customized data bases, organizing and publishing research results . . . even providing videos.

For more information about the NASA STI Program Office, see the following:

- Access the NASA STI Program Home Page at ***http://www.sti.nasa.gov***

- E-mail your question via the Internet to help@sti.nasa.gov

- Fax your question to the NASA Access Help Desk at 301–621–0134

- Telephone the NASA Access Help Desk at 301–621–0390

- Write to:
  NASA Access Help Desk
  NASA Center for AeroSpace Information
  7121 Standard Drive
  Hanover, MD 21076

NASA/CR—2001-210709

Mars Flyer Rocket Propulsion Risk Assessment

ARC Testing

Atlantic Research Corporation
Niagara Falls, New York

Prepared under Contract NAS3–99197

National Aeronautics and
Space Administration

Glenn Research Center

April 2001

Trade names or manufacturers' names are used in this report for identification only. This usage does not constitute an official endorsement, either expressed or implied, by the National Aeronautics and Space Administration.

Available from

NASA Center for Aerospace Information
7121 Standard Drive
Hanover, MD 21076
Price Code: A04

National Technical Information Service
5285 Port Royal Road
Springfield, VA 22100
Price Code: A04

Available electronically at http://gltrs.grc.nasa.gov/GLTRS

TABLE OF CONTENTS

LIST OF TABLES

TABLE OF CONTENTS (Continued)

LIST OF FIGURES

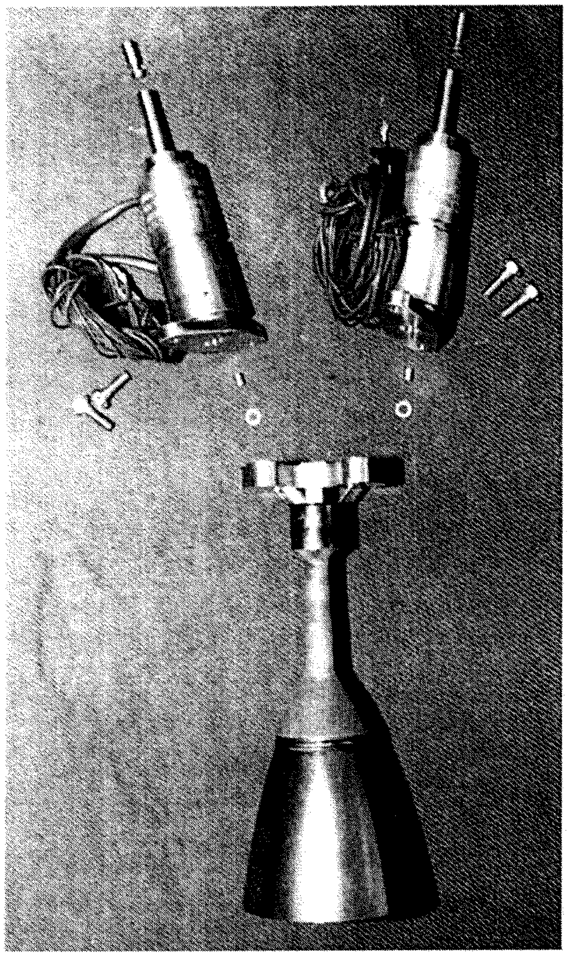
1.0 INTRODUCTION

The purpose of this report is to describe the results from tests conducted at the Atlantic Research Corporation/Liquid Prolusion Division (ARC/LP) located in Niagara Falls, NY under the NASA Glen Mars Flyer Rocket Propulsion Risk Assessment Program. The technical objectives of this program were to provide test data on the operational characteristics of a 2 lbf thruster operating with MMH and MON-25 propellants cooled to –40C to simulate conditions expected to be encountered during the Mars Flyer mission.

The thruster used in this program was an ARC 10N (2.25 lbf) thruster which had been used in previous ARC development activities. The thruster was made available to this program at no cost and is described in Section 2.0. Since MON-25 is not available off-the-shelf, ARC manufactured a supply of MON-25 for this program and verified the NO content by assay using an independent company, Vicksburg Chemical. The procedure used to manufacture the MON-25 and the assay results are discussed in Section 3.0. For this program, ARC developed a propellant conditioning system which was capable of delivering –40C propellant to the thruster for tests of any duration. A description of this system is given in Section 4.0. The test plan is discussed in Section 5.0 and the results from the tests are discussed in Section 6.0. Conclusions are given in Section 7.0 and Recommendations in Section 8.0

This program can be summarized by indicating the manufacture of MON-25 was successful with two assays indicating the NO content of the propellant was 25.0 and 25.3%. After a series of development and calibration tests, the propellant conditioning system demonstrated the capability of delivering propellant to the thruster at the target temperatures of 21C (70F), -1C (30F), -18C (0F), -29C (-20F) and –40C (-40F) for tests of any run length, including three successive tests of 1200s duration of –40C. All but three of the tests in the original plan were conducted with testing being terminated with the depletion of the MON-25. Most tests were successfully completed including two where the test cell pressure was increased to 10 torr (0.2 psia) to simulate the Martian atmospheric pressure. The thruster ran well at

Figure 1. ARC 2 lbf Thruster



Design

- Propellants: MON-3/MMH
- Materials
- Disilicide Coated C-103 Chamber
- Titanium Injector & Cone
- Area Ratio: 336/1
- Mass: 0.62 kg
- Status
- Flight Qualified
- 210 Delivered
- 50 On-Orbit

Performance

- Thrust: 9.4N
- Feed Pressure: 13.8 bar
- Mixture Ratio: 1.65
- Specific Impulse: 274s
- Chamber Temperature: 900C
- Demonstrated Life
- 151,000s
- 587 kg throughput
- 1278 Cold Starts
- 1,120,000 Pulses

most conditions with the most notable result being that the cold propellant caused the nominal mixture ratio to shift from 1.65 to 1.90 as the propellant temperatures were reduced from 21C to –40C with a consequent small decrease in specific impulse. Details of the test results are given in Section 6.0 and Volume 2. These results show that there should be no significant problems operating this thruster with MMH/MON-25 propellants in the cold –40C environment.

2.0 THRUSTER DESCRIPTION

The 10N thruster, which was designed for MON-3/MMH propellants, is shown in Figure 1. The thruster is radiation cooled and uses fuel-barrier cooling to maintain chamber wall temperatures at levels sufficiently low to ensure the thruster meets propellant throughput requirements. The thruster was designed for stationkeeping on geostationary satellites and has been fully qualified for this mission using solenoid valves.

The thruster was designed for operation with propellants at a nominal temperature of 21C (70F). Orifices and flow passages were designed to provide a nominal mixture ratio of 1.65 with MMH and MON-3 propellants at 21C. Since propellant densities and viscosities are a function of propellant temperature, it was expected that operation with –40C propellants would result in changes in the thruster operational characteristics. These changes will be discussed shortly. It is important to note that the purpose of the tests described herein was to obtain an understanding of thruster operation with –40C propellants such that one could design a thruster optimized for these conditions at a later time.

The thruster is comprised of three major sub-assemblies: the injector assembly, thrust chamber and expansion cone and the propellant valves. The injector assembly consisting of the injector, distribution ring, thermal stand-off and oxidizer inlet tube are machined from 6Al/4V Titanium and electron beam welded together. The injector has three unlike doublets in the core region and six fuel film coolant holes equispaced on the periphery. Gold plated Inconel 'C' seals provide the sealing between the valves and the injector assembly..

The thrust chamber is manufactured from Columbium C103 alloy and is coated with the Hitemco R512E disilicide coating. The 336:1 area ratio expansion cone is machined from 6Al/4V Titanium and is electron beam welded to the thrust chamber.

This thruster has no Pc top so chamber pressures could not be measured. Chamber pressure was estimated by using measured thrust and an assumed value of the thrust coefficient Cf of 1.77.

The thruster uses two Moog Model 51-178 series redundant normally closed solenoid valves. The valve is a fail safe design that remains closed through spring pre-load until opened by electrical energisation. The valve features a teflon seat and no sliding fits. An outlet orifice is fitted into the valve flange for pressure drop control and there is a provision for an orifice in the inlet tube for final thruster trimming. The valve also incorporates a 25 micron absolute filter at the inlet.

2.1 Thruster Heritage

The thruster has been subjected to a very extensive qualification program using two thrusters. One thruster was used primarily for steady-state testing, while the second was used primarily for pulse mode testing. The scope of the qualification program included:

- Two thrusters
- 1400 burns
- Propellant temperatures: -7 to 54C
- Feed pressures: 8.3 – 19.8 bar
- Pulse trains of over 2700 pulses
- Hot restarts
- Variable pulse width pulse trains
- Thermal stability tests
- Gas Ingestion Tests
- Thermal Soakback Tests

Some results on the qualification program scope are included in Table 1. The thruster has successfully performed long burns of 1000s and it was expected that the thruster would be able to perform the 20 minute (1200s) burns required during this program without any problems.

Table 1. Qualification Test Summary

Parameter	Results	
	Qual Thruster 1	**Qual Thruster 2**
Total number of pulses	1,123,977	216,863
Thermal cycles (cold restarts)	1,278	426
Propellant throughput • Steady State • Pulse Mode	 417 877	 645 154
Propellant temperature range (C)	-10 to 59	-9 to 57
Injector pre-fire temp range (C)	-20 to 96	-11 to 97
Mixture Ratio	1.08 to 1.96	1.31 to 1.97
Thrust (N)	5.8 to 12.5	5.8 to 12.0
Longest SSF duration (s)	1,000	800
Longest PMF duration (s)	3,600	812
Total on time (s)	151,362	99,354
Total Impulse (N-s)	1,428,192	1,030,620

Figure 2 shows the fuel and oxidizer feed pressure envelope for which the thruster is qualified and the points at which tests were conducted. The qualification range was 8.3 – 19.7 bar (120-286 psia). The nominal operating conditions for the thruster are feed pressures for the fuel and oxidizer of 13.8 bar (200 psia) and a mixture ratio of 1.65. ARC plans to put trim orifices into the thruster which will result in a nominal feed pressure of 15.2 bar (220 psia).

Figure 2. Thruster Feed Pressure Qualification Box

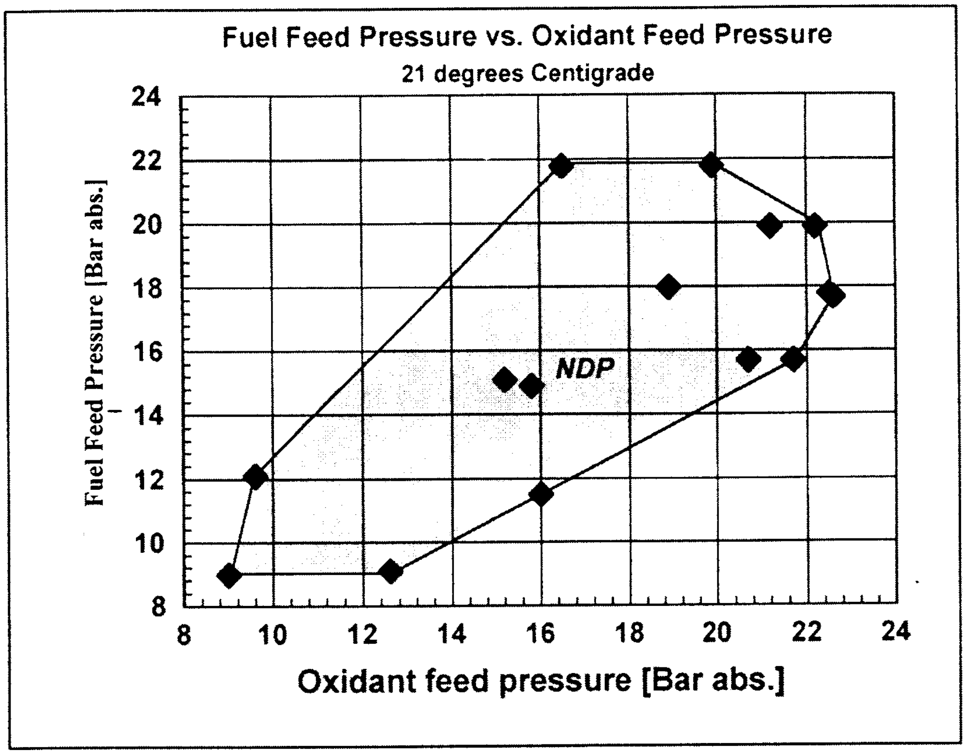


Figure 3 shows a map of the pulse mode tests which were conducted on the thruster in terms of the on-time and off-time for the pulses and the propellant temperatures at which these tests were conducted. The pulse mapping on this thruster was quite extensive and included long pulse trains (up to 4000 pulses) to demonstrate thermal stability. The thruster demonstrated successful operation at all conditions and no thermal limitations were discovered during the qualification tests.

Figure 3. Pulse Mode Operation Box

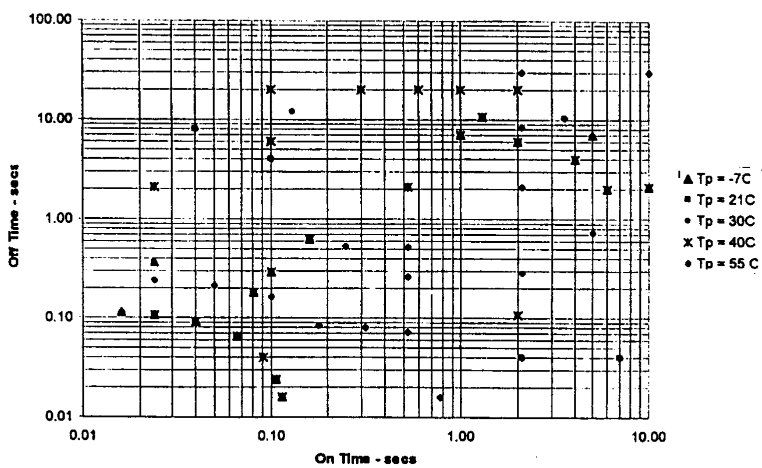
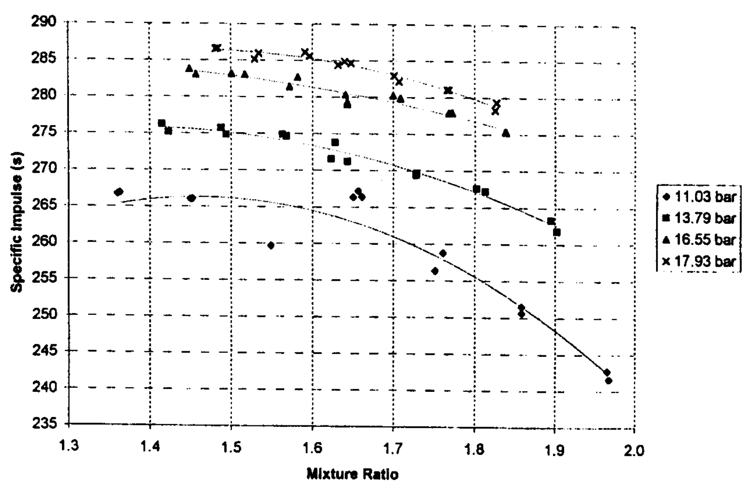


Figure 1 provides a brief summary of the operating characteristics of the thruster at the nominal operating condition. The specific impulse is 274s and the maximum chamber temperature is only 900C (1650F). This thruster was designed with a high fraction of the fuel in the barrier in order to produce a thermally robust thruster which could satisfy all high propellant throughput requirements for very demanding satellite missions. The thruster was designed to operate at a mixture ratio of 1.65 with MON-3/MMH propellants.

The thruster was designed to provide moderate performance in terms of Isp, but, more importantly, to operate without any thermal limitations over the entire range of duty cycles which could be encountered during satellite operation. The nominal mixture ratio for the thruster is 1.65 with MON-3/MMH and the design has a high fraction of the fuel in the barrier. At nominal conditions, the Rupe Number for the doublet element core is about 1.75, rather than the more optimum value of 1.00; this is by design to obtain a wide operational envelope rather than a narrow envelope with higher performance. Rupe Number decreases toward a more optimum value of 1.0 as mixture ratio decreases below 1.65 and Isp increases slowly as MR decreases as shown by the data in Figure 4.

Figure 4. Variation of Specific Impulse with Mixture Ratio



2.2 MARS FLYER TEST CONDITIONS

As mentioned earlier, the thruster was designed for operation with 21C propellants and some changes in operational characteristics were expected at the Mars Flyer conditions due to the change in propellant properties. The propellant properties which have the largest influence on thruster operation are density and viscosity.

Figure 5 shows how the density of MON-3, MON-25 and MMH vary with temperature over the –46C (–50F) to 49C (120F) range. All three propellants show the same trend of density increasing slowly as propellant temperature decreases. Note that MON-25 is somewhat less dense than MON-3 due to the higher NO content.

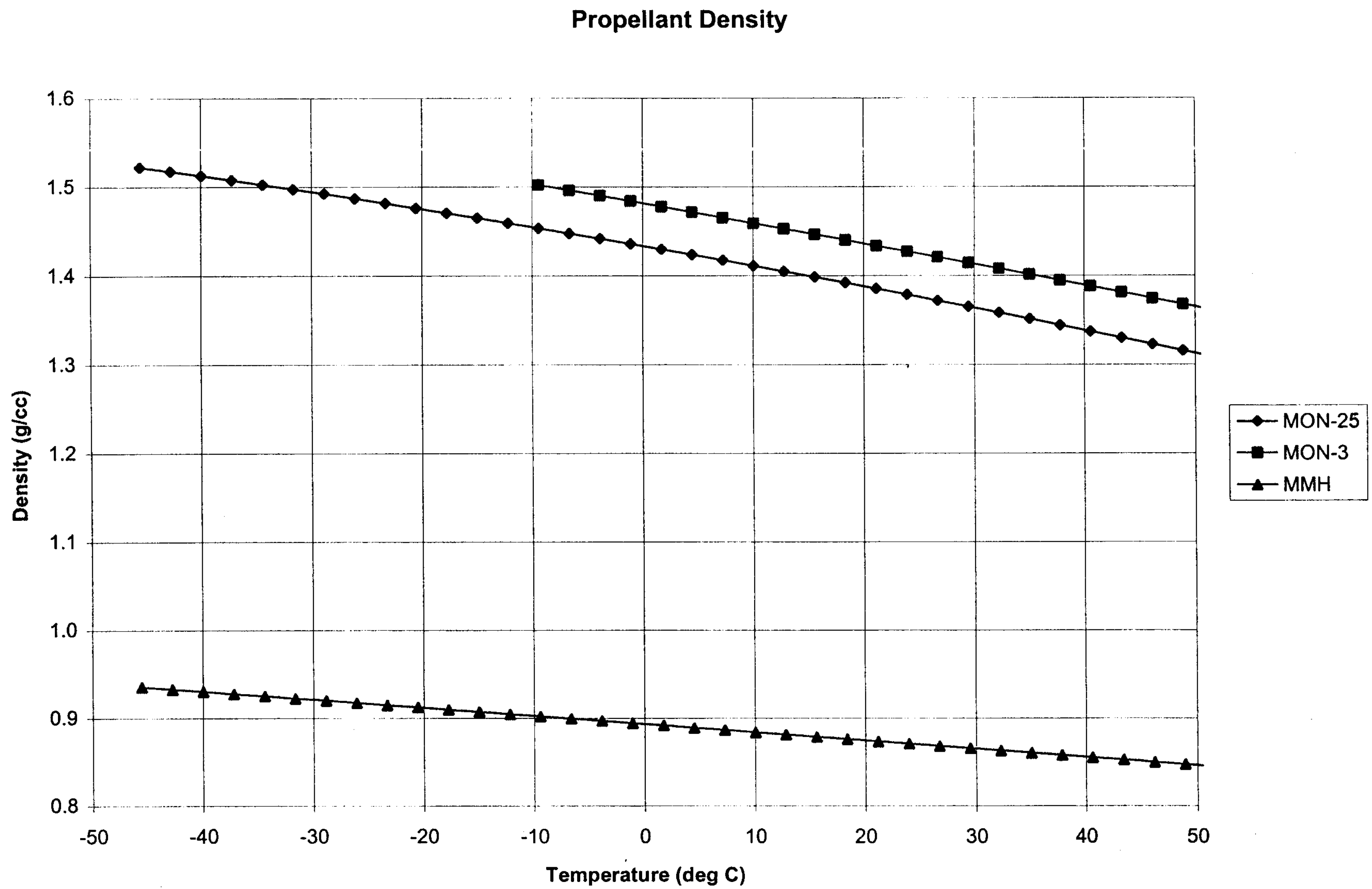


Figure 5. MON-3, MON-25, and MMH Density Variation with Temperature

In assessing performance of bipropellant thrusters, one of the most common design parameters is the Rupe Number given by

$$R = \frac{(\rho\, u^2 d)\, ox}{(\rho\, u^2 d)\, fuel} \tag{1}$$

In Eq. (1), ρ is the propellant density, u is the orifice injection velocity and d is the orifice diameter; these parameters are for the injector core. Optimum bipropellant combustion is obtained for R = 1.0. Eq. (1) can be rewritten as

$$R = \left(\frac{\rho_f}{\rho_o}\right)\left(\frac{MR}{1-B}\right)^2\left(\frac{d_f}{d_o}\right)^3 \tag{2}$$

where ρ_f and ρ_o are the fuel and oxidizer densities, d_f and d_o are the diameters of the fuel and oxidizer orifices in the injector core, MR is the mixture ratio, and B is the fraction of the barrier fuel injected into the thrust chamber wall. The parameter B is fixed by the ratio of the fuel core and barrier orifice diameters. Thus, for a given thruster design, R is affected by test conditions only through the density ratio and mixture ratio.

Figure 6 shows the oxidizer/fuel density ratio for MON-3/MMH and MON-25/MMH as a function of temperature. For MON-3/MMH at 21C, the density ratio is approximately 1.64 while for MON-25/MMH at –40C, the value is about 1.625. The difference between these two values in terms of their effect on Rupe Number is negligible. In fact, the variation of the density ratio for both propellant combinations over the range of temperatures shown in Figure 6 is sufficiently small that no significant effect on Rupe Number or thruster performance would be expected. The conclusion then, is that the thruster performance will not be affected to any significant degree by the propellant density changes associated with operating at –40C.

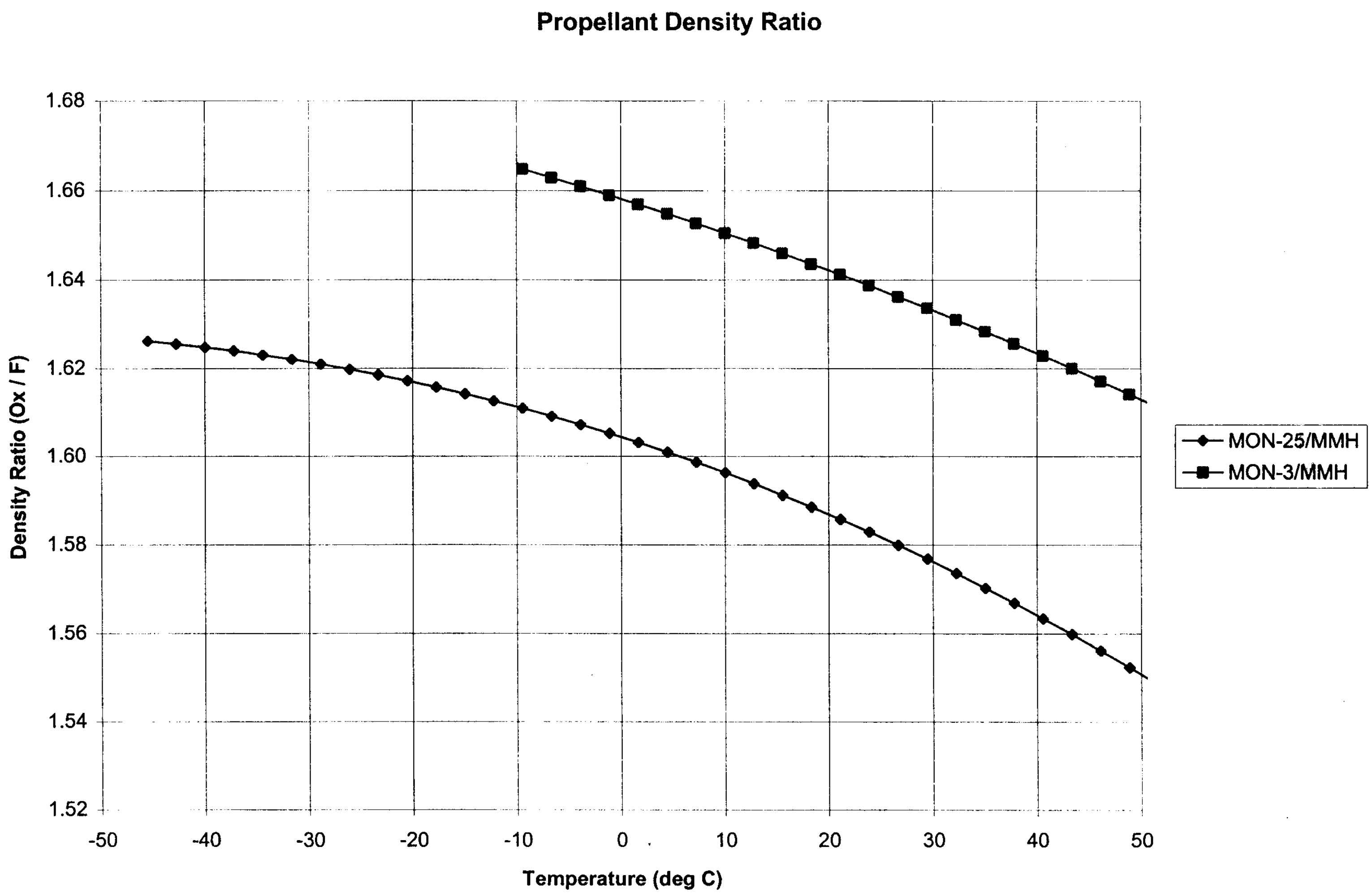


Figure 6. Fuel/Oxidizer Density Ratios

Figures 7 and 8 show the variation of viscosity of MMH and MON-3 with temperature. Note that the MON-3 data extends downward only to 2C (35F). No data is available for the viscosity of MON-25 so we will use the MON-3 data as an estimate. The data in Figure 7 show that in going from 21C to –40C, the viscosity of MMH increases by about a factor of 7. Using a straight line extrapolation of the data in Figure 8, in going from 21C to –40C, the viscosity of MON-25 would be estimated to increase by about a factor of 2.

For a thruster operating at a given feed pressure, an increase in propellant viscosity will result in an increase in the frictional pressure drops losses through the thruster flow passages and consequently, a reduction in propellant flowrate. Since the increase in the fuel viscosity as propellant temperatures are reduced to –40C is much larger than the increase in the oxidizer viscosity, the fuel flowrate will decrease more than the oxidizer flowrate. This will cause the nominal thruster mixture ratio to increase as propellant temperature decreases and as will be shown in Section 6, this is the trend that is observed in the test data.

The results in Figure 4 show that thruster Isp decreases slowly as mixture ratio increases and therefore, we would expect Isp to decrease as propellant temperature decreases because of the mixture ratio shift. This trend is also observed in the test data.

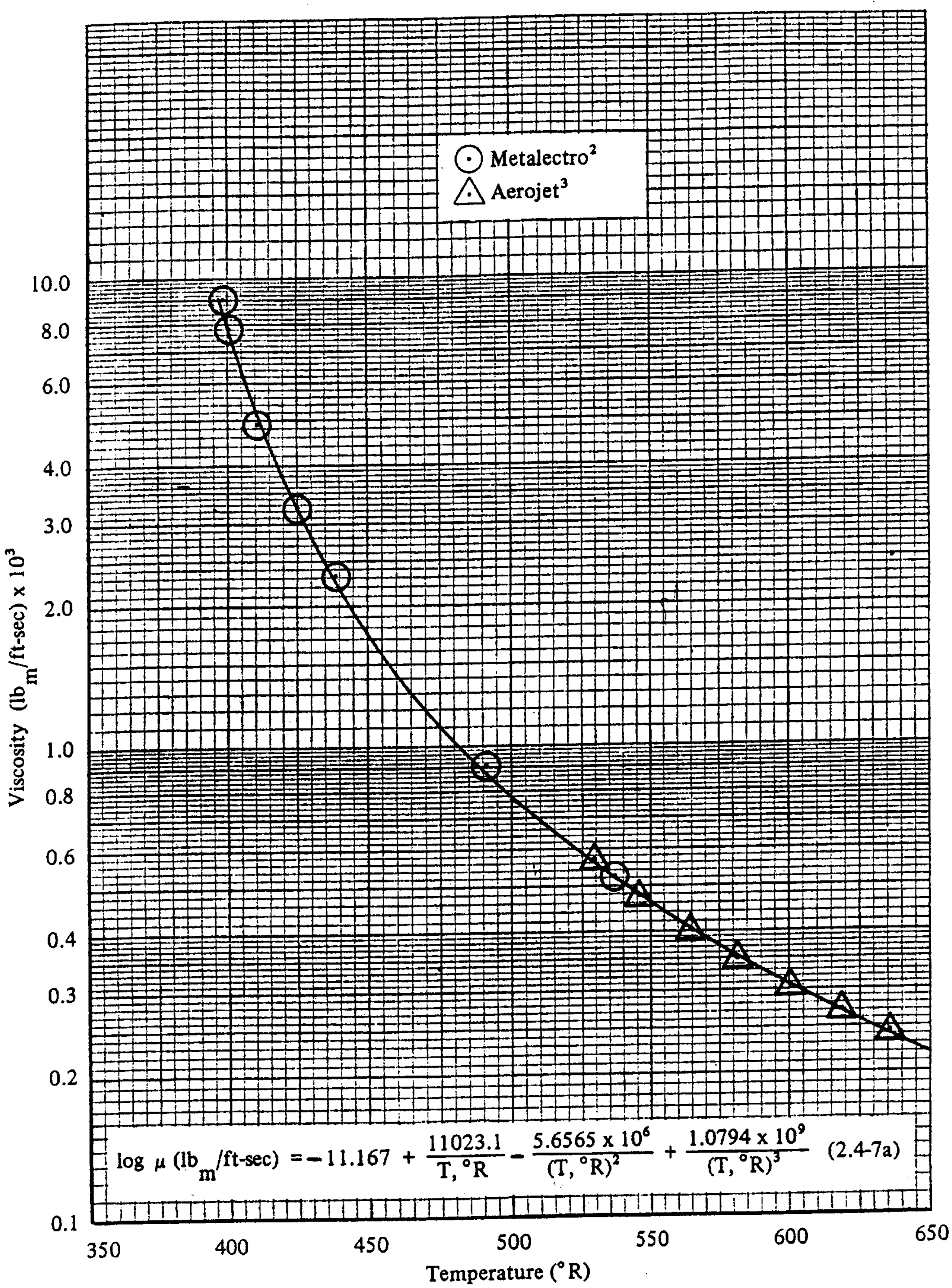


Figure 7. Variation of MMH Viscosity with Temperature

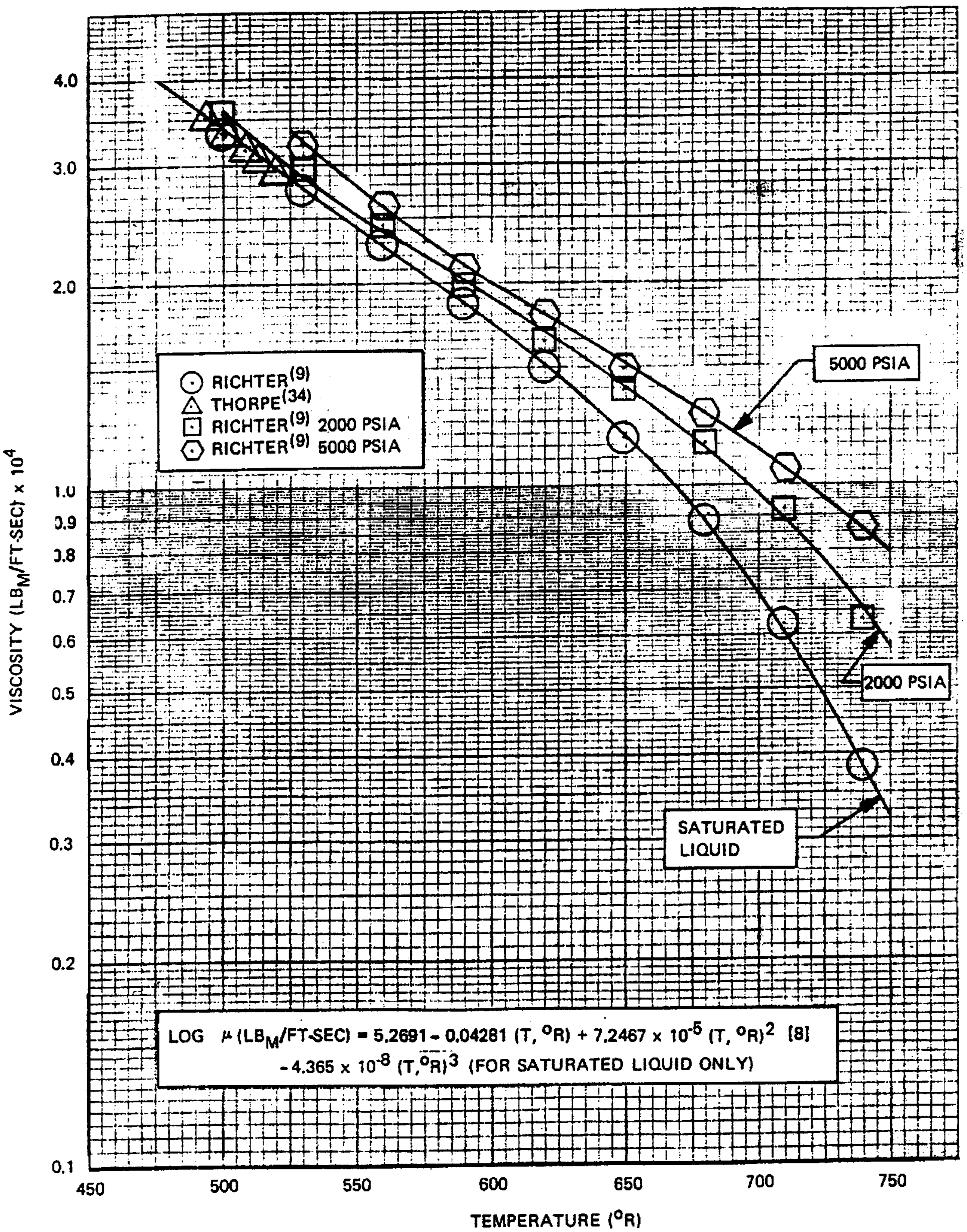


Figure 8. Variation of MON-3 Viscosity with Temperature

3.0 MON-25 PRODUCTION PROCESS

MON-25 is not available either commercially or from Kelly AFB stockpiles. Therefore, the MON-25 used in this program was manufactured by ARC/LP at our Niagara Falls facility. ARC/LP had experience in the manufacturing of MON-3 and MON-10 from previous and ongoing programs. The technique is straightforward as long as proper handling and mixing procedures are followed.

MON-25 consists of a mixture of 75% nitrogen tetroxide (N2O4) and 25% nitric oxide (NO) by weight. The starting constituents for the mixture are MON-3, which is readily available at our facility from other programs, and commercially available bottles of NO. The mixing process involves an exothermic reaction of the NO gas with N2O4 to create nitrogen trioxide (N2O3), which gives MON-25 its characteristic bluish-green color. ARC employed a cooling chamber during the mixing operation to both limit the oxidizer temperature increase from the reaction and to lower the vapor pressure of the resulting mixture to better utilize the available NO from its storage bottle. The initial temperature of the mixing bottle and MON-3 was approximately –1C (30°F).

Figure 9 shows a schematic of the mixing operation. All hardware associated with the production of MON-25 was passivated to ensure the minimization of MON-25 leaching excess iron into solution. This included all mixing vessels, transfer/feed lines and samples bottles. To minimize the transfer operations required for this test series, a common mixing/storage vessels was used. The mixing/storage vessel consisted of a standard cylindrical 9-gallon tank with a stand tube inside along its center axis. The gaseous NO is introduced into the tank through this perforated stand tube at the bottom of the tank.

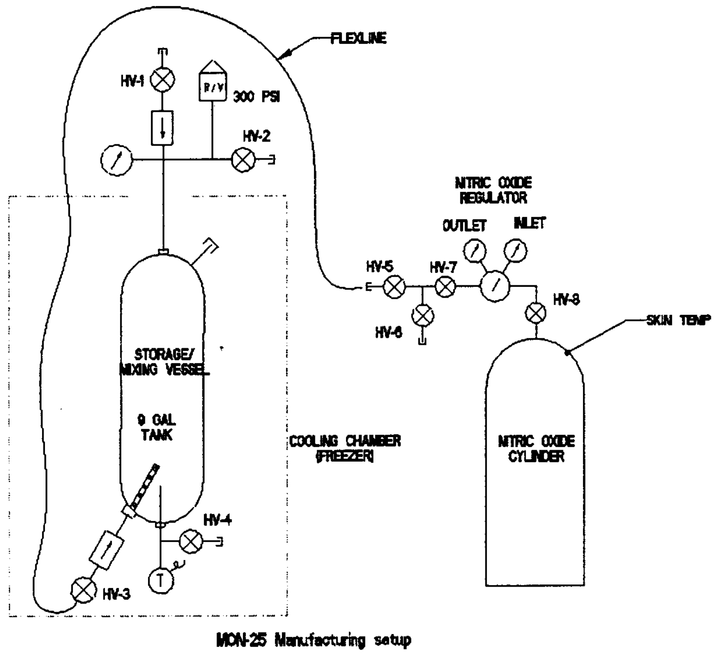


Figure 9. MON-25 Mixing Operation Schematic

The flow of NO into the tank was controlled by a NO gas regulator which limited the pressure of the NO to about 1.6 bar (23 psia), which is approximately the vapor pressure of the final mixture of MON-25 at –1C (30F). Both pressure and temperature were monitored during the mixing operation. Periodically, the NO flow was turned off to allow the reaction to reach completion and for the mixing/storage vessel to be recooled. The pressure and temperature of the NO bottles were monitored during this operation. The NO bottle was allowed to return to ambient temperature to calculate the mass of NO which has been introduced into the mixing/storage vessel through the use of the perfect gas law.

After it has been determined that sufficient NO has been added to the oxidizer, the oxidizer was sampled using an evacuated standard Hoke bottle. The sample was shipped to Vicksburg Chemical for chemical analysis per MIL-P-26539D to verify the NO content of the oxidizer. The two samples analyzed showed an NO content of 25.0% and 25.3%. Approximately 22.7 kg (50 lbm) of MON-25 was manufactured.

After completion of the mixing operation, gaseous helium was introduced into the mixing/storage vessel. This gas pressure blanket insures that the NO does not come out of solution preferentially at ambient temperatures. The pressure was maintained at about 8.6 bar (125 psia), which is higher than the vapor pressure of MON-25 at 32C (90F) (a maximum summer time temperature for the Niagara Falls facility).

4.0 PROPELLANT AND HARDWARE TEMPERATURE CONDITIONING SYSTEMS

To support the Mars Flyer program, ARC developed a propellant conditioning system with the capability to cool the propellants to –40C for tests of any run duration. A schematic of the propellant cooling system is shown in Figure 10.

In this system, liquid carbon dioxide (CO_2) is used to cool a bath filled with a mixture of approximately 60/40 ethylene glycol and water. The freezing point for this mixture is about –50C (-58F). This mixture was chosen as a tradeoff between the required temperature capability of the fluid and the ability to pump it to the propellant heat exchangers. For example, by reducing the water in the fluid, the freezing point could be reduced further, however, since the viscosity of the fluid increases rapidly as the ethylene glycol content is increased and temperature is reduced, it becomes difficult to pump an adequate supply of the fluid to the heat exchangers. After some experimentation, the 60/40 mixture was found to be satisfactory for these tests. The CO_2 circulates through the bath inside a set of copper coil tubing. It enters the coils as a liquid and exits as a gas; most of the cooling capability is, therefore, in the heat of vaporization of the CO_2. A mechanical agitator was also used in the bath to increase the heat transfer rate between the bath fluid and the liquid CO_2.

The glycol bath is located outside of the test cell. The cold bath fluid is pumped through an insulated line to the propellant heat exchanges inside the test cell. The cooling fluid is split inside the cell and directed to separate fuel and oxidizer heat exchangers. After leaving the propellant heat exchangers, the fluid is returned to the bath via a single line.

As shown in Figure 10, the propellant heat exchangers are located in the test cell downstream of the propellant tanks and flowmeters. Figure 11 shows the thruster mounted in the test cell with the propellant heat exchangers located just behind the thruster. The distance between the propellant heat exchangers and the thruster valve was approximately 10 cm. Thermocouples (FIT, OIT; See Figure 10) were located in the fuel and oxidizer lines just

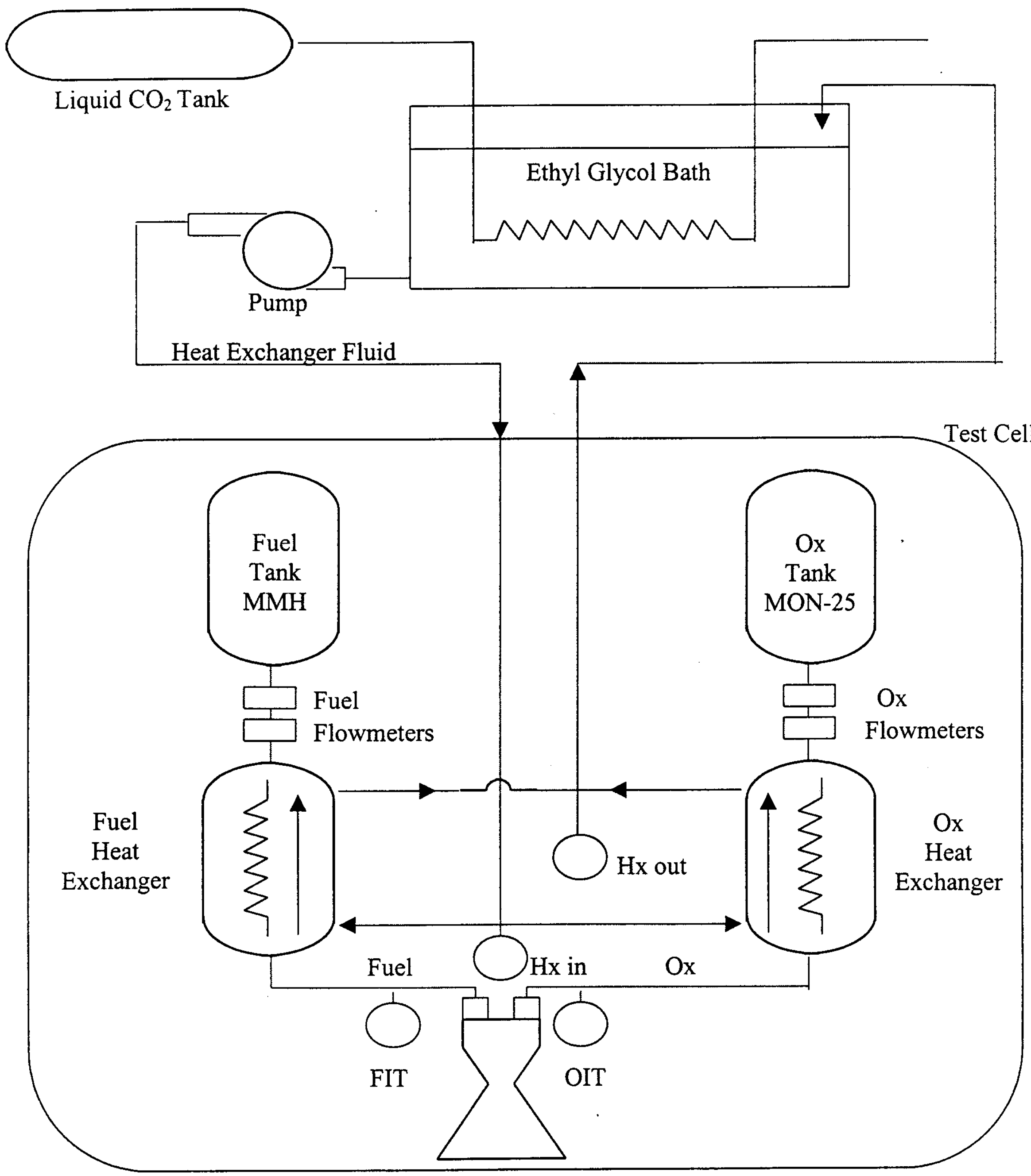


Figure 10. Propellant Cooling System in Test Cell

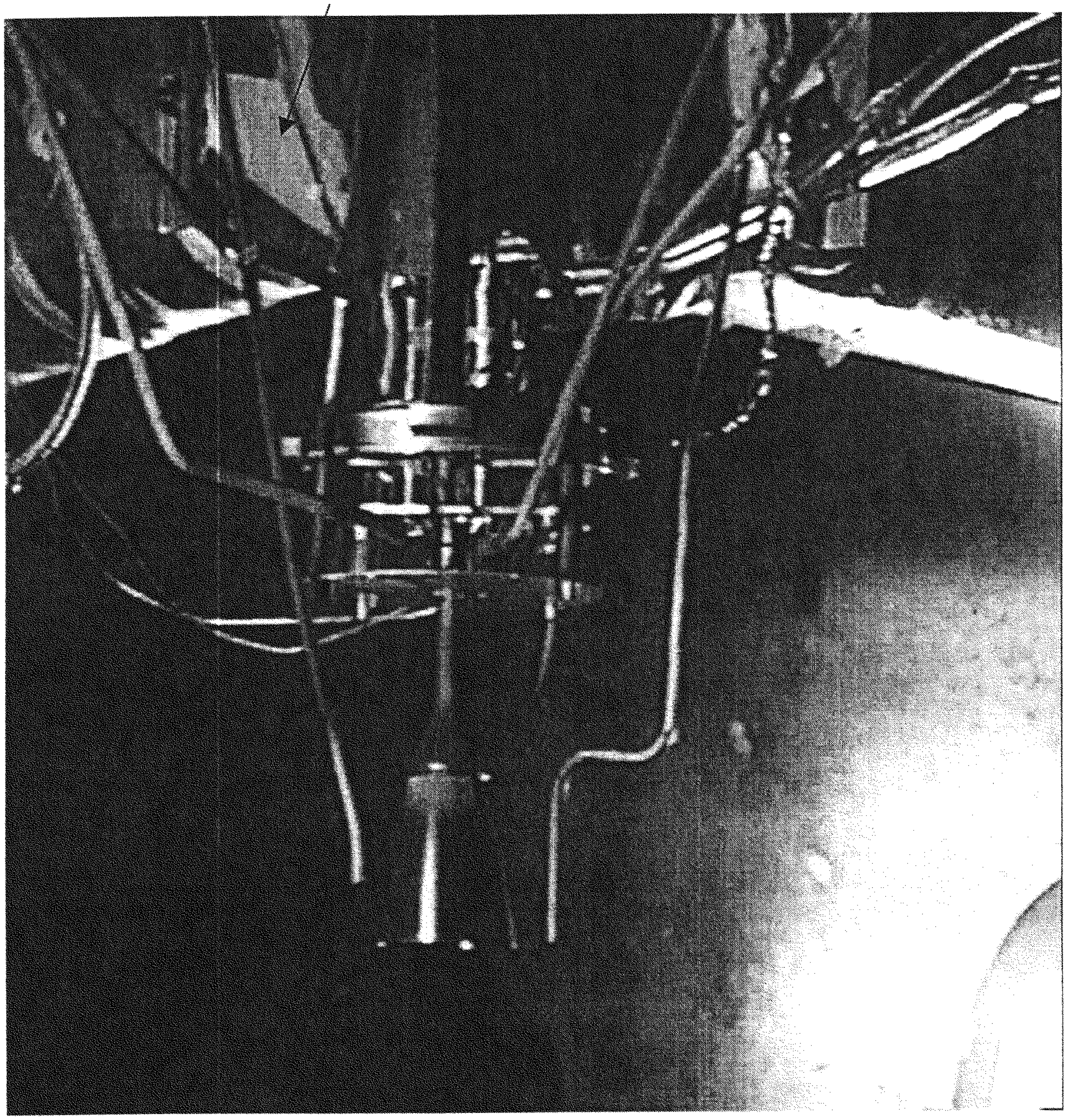


Figure 11. Thruster Mounted in Test Cell

upstream of the thruster to measure propellants temperatures as close as possible to the thruster inlet. The propellant heat exchangers contain copper coils through which the propellants flow. The cold glycol/water mixture flows around the outside of the coils and cools the propellant. The heat exchanger is a counterflow design with the cold glycol entering the heat exchanger at the location where the propellant exits to the thruster. Typically, the propellant enters the heat exchanger at about 21C (70F) and leaves at a temperature within about 1C (2F) of the entering glycol/water mixture.

Figure 12 shows a high resolution time history of the propellant temperatures for Test 35625 which was a 1200s run at the nominal 15.2 bar (220 psia) feed pressure with nominal –40C propellants. Except for a short period during the start transient, the propellant conditioning system maintained the propellants within ±1C (±2F) of 40C for the entire 1200s run. The start transient is due to the cleaning of the small amount of propellant in the short line between the propellant heat exchangers and the thruster. Before each test, a short bleed flow is conducted into a catch tank to clear the propellant in this line, however, bleed flow times on these tests was kept short due to the limited amount of MON-25 available for these tests.

As noted earlier, the mixture of ethylene glycol and water used as a heat exchanger fluid becomes very viscous at low temperatures. When conditioning propellants to –40C, it was found that the heavy load imposed on the pump by this viscous fluid would cause the pump to over heat and cycle off during long tests. The two valleys in the propellant temperatures shown in Figure 12 are due to the pump cycling off and then being restarted. Figure 13 shows the time history of temperatures of the heat exchanger fluid at the heat exchanger inlet and outlet for this test. These temperatures are quite constant over most of the test except for the two points where the pump cycles off and is restarted. Note that in cooling the propellant, the temperature increase of the fluid is only 1C (2F) from heat exchanger inlet to exit. The system was designed with a good deal of margin in terms of the thermal cooling capability of the propellant heat exchangers and because of this margin, even when the pump shuts down and is restarted, the propellant temperatures were maintained within 1C of the target –40C valve for the entire 1200s test.

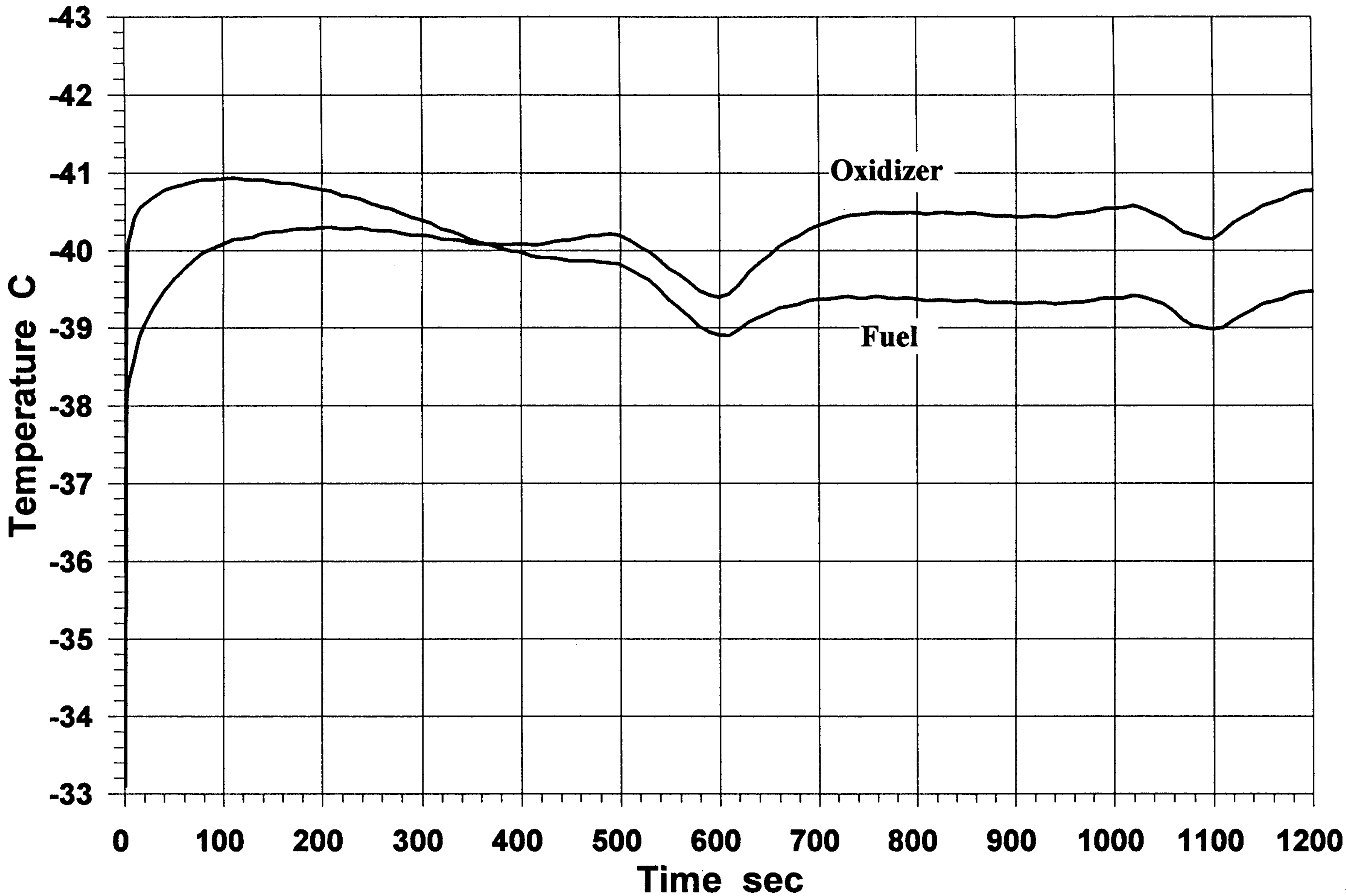


Figure 12. Propellant Temperature Time History, Test 35625

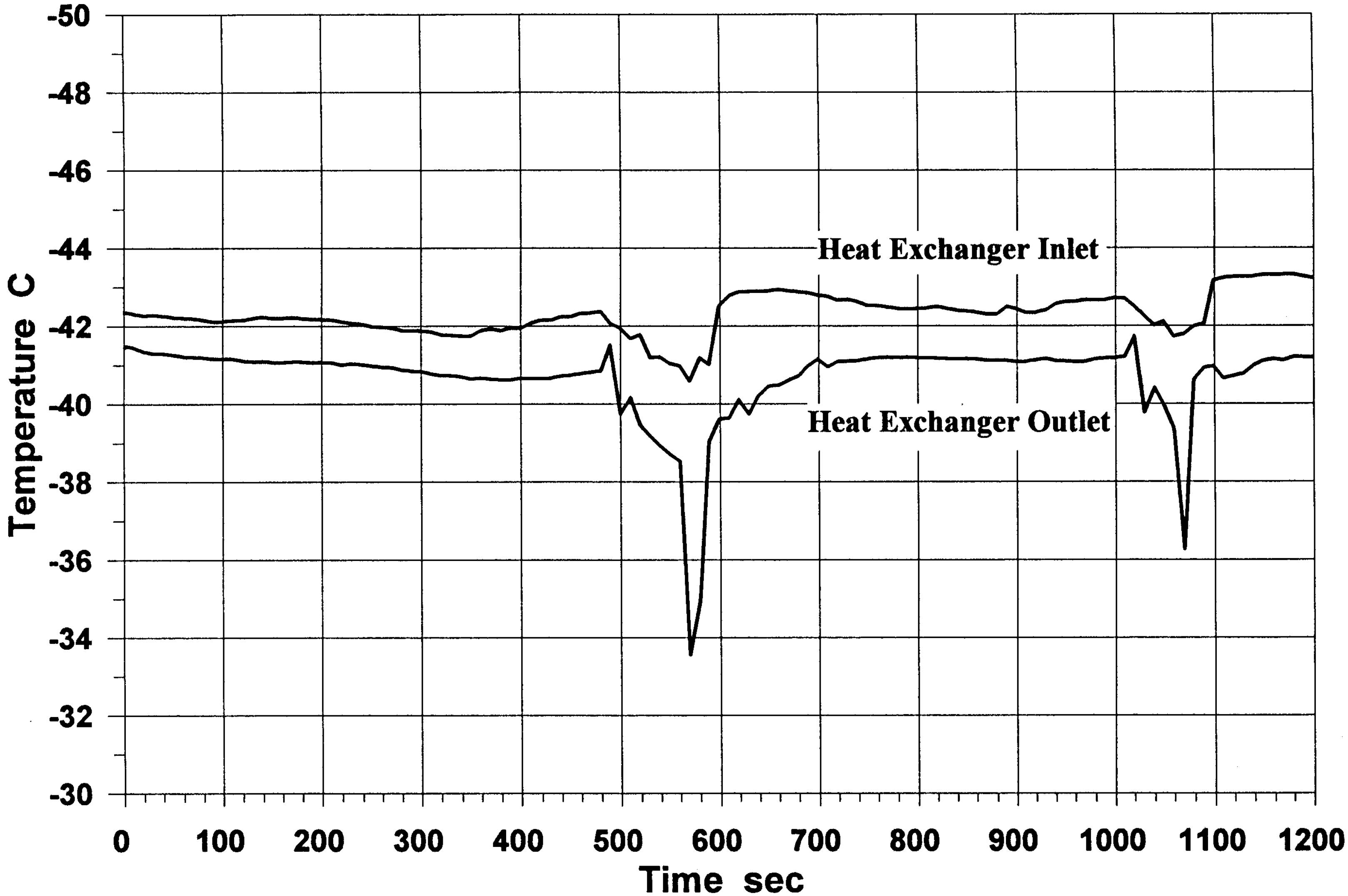


Figure 13. Heat Exchanger Inlet and Outlet Temperatures, Test 35625

Before each test, the thruster and valves were cooled to a temperature which was below the propellant temperature. It was difficult to cool the hardware with any degree of precision, however, the objective was hardware temperatures equal to or lower than the propellant temperatures, and this was achieved in all cases. Hardware temperatures were measured by thermocouples located on the back of the injector. The hardware was cooled by throttling liquid CO_2 through an orifice. In going through the orifice, the liquid flashed to vapor at the test cell pressure of about 1.5 torr (0.03 psia). This cold CO_2 gas was directed at the hardware through four lines, one at each valve and two at the injector from opposing sides. This cold gas injection system had independent valves for the valve and injector cooling circuits and they were cycled on/off until the desired temperatures were obtained. As noted earlier, it was difficult to achieve precise control of the hardware temperatures using this approach. The data shows that for all tests, hardware temperatures were below the propellant temperatures and for many tests, the hardware start temperatures were well below the propellant temperatures which represents a very severe environment. In any future testing, a system which provides more precise control of hardware temperatures should be developed.

5.0 TEST PLAN

The test procedures and test plan for this activity are documented in ARC PP-80704 (Test Procedure) and ARC PP-80704, Addendum 1 (Hot Fire Test Matrix). These documents were reviewed and approved by NASA before testing was initiated.

Tables 2, 3 and 4 show the planned tests. Table 2 shows the baseline tests with 21C (70F) propellants. The test matrix consists of tests a different feed pressures from 6.9 – 20.7 bar (100-300 psia), three tests to measure the effect of mixture ratio, and one test where the cell pressure was increased to 10 torr (0.2 psia) to simulate the Mars atmospheric environment. Shown in the last column is the ARC test number for the planned test. Table 3 shows the matrix for the tests to map thruster performance as a function of propellant temperature and feed pressure. Table 4 shows the matrix for testing with –40C propellants. All tests were completed except for Tests 6, 8, and 9 in Table 4. These tests were not conducted since of the MON-25 supply was exhausted.

Figure 14 shows the thermocouple instrumentation on the thruster. Two thermocouples (ICBT1, ICBT2) wee located on the backside of the injector. One thermocouple was located on each valve and valve mount and four thermocouples were located at the nozzle joint. These thermocouples were used to set pre-test hardware temperature and to monitor thruster behavior during hot fire testing.

Table 2
MON-25/MMH Baseline Test Matrix

Test No.	FFP/OFP (psia)	Run Duration: On/Off (sec)	Run Duration: No. of Pulses	Temperatures (^oF): Propellant	Temperatures (^oF): Start Valve	Temperatures (^oF): Start Inj.	Flow Path	Comments	ARC Test Number
1	220/220	60 / -	1	70	≤104	≤104	Flowmeter	Baseline	35606
2	100/100	60 / -	1	70	≤104	≤104	Flowmeter	Pc Survey	35607, 35610
3	150/150	60 / -	1	70	≤104	≤104	Flowmeter	Pc Survey	35601
4	200/200	60 / -	1	70	≤104	≤104	Flowmeter	Pc Survey	35608
5	250/250	60 / -	1	70	≤104	≤104	Flowmeter	Pc Survey	35611
6	300/300	60 / -	1	70	≤104	≤104	Flowmeter	Pc Survey	35612
7	220/250	60 / -	1	70	≤104	≤104	Flowmeter	MR Survey	35613
8	220/190	60 / -	1	70	≤104	≤104	Flowmeter	MR Survey	35614
9	190/220	60 / -	1	70	≤104	≤104	Flowmeter	MR Survey	35615
10	220/220	0.1/0.1	20	70	≤104	≤104	Bellows	Baseline Pulse	35616
11*	220/220	60/-	1	70	≤104	≤104	Flowmeter	Mars Simulation	35617

Table 3.
MON-25/MMH Pressure and Temperature Mapping Tests

Test No.	FFP/OFP (psia)	Run Duration		Temperatures (°F)			Flow Path	Comments	ARC Test Number
		On/Off (sec)	No. of Pulses	Propellant	Start				
					Valve	Inj.			
1	100/100	60 / -	1	-40	≤-40	≤-40	Flowmeter	Pc Survey	35619
2	150/150	60 / -	1	-40	≤-40	≤-40	Flowmeter	Pc Survey	35620
3*	220/220	60 / -	1	-40	≤40	≤40	Flowmeter	Mars Simulation	35621
4	250/250	60 / -	1	-40	≤-40	≤-40	Flowmeter	Pc Survey	35622
5	300/300	60 / -	1	-40	≤-40	≤-40	Flowmeter	Pc Survey	35623
6	100/100	30 / -	1	-20	≤-20	≤-20	Flowmeter	Pc Survey	36639
7	150/150	30 / -	1	-20	≤-20	≤-20	Flowmeter	Pc Survey	35640
8	220/220	60 / -	1	-20	≤-20	≤-20	Flowmeter	Pc Survey	35641
9	250/250	30 / -	1	-20	≤-20	≤-20	Flowmeter	Pc Survey	35642
10	300/300	30 / -	1	-20	≤-20	≤-20	Flowmeter	Pc Survey	35643
11	100/100	30 / -	1	0	≤0	≤0	Flowmeter	Pc Survey	35634
12	150/150	30 / -	1	0	≤0	≤0	Flowmeter	Pc Survey	35635
13	220/220	60 / -	1	0	≤0	≤0	Flowmeter	Pc Survey	35636
14	250/250	30 / -	1	0	≤0	≤0	Flowmeter	Pc Survey	35363
15	300/300	30 / -	1	0	≤0	≤0	Flowmeter	Pc Survey	35638
16	100/100	30 / -	1	30	≤30	≤30	Flowmeter	Pc Survey	35629
17	150/150	30 / -	1	30	≤30	≤30	Flowmeter	Pc Survey	35630
18	220/220	60 / -	1	30	≤30	≤30	Flowmeter	Pc Survey	35630
19	250/250	30 / -	1	30	≤30	≤30	Flowmeter	Pc Survey	35632
20	300/300	30 / -	1	30	≤30	≤30	Flowmeter	Pc Survey	35633

Table 4.
MON-25/MMH Test Matrix at –40C

Test No.	FFP/OFP (psia)	Run Duration On/Off (sec)	Run Duration No. of Pulses	Temperatures (°F) Propellant	Temperatures (°F) Start Valve	Temperatures (°F) Start Inj.	Flow Path	Comments	ARC Test Number
1	220/220	1200 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	35625
2	220/220	1200 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	35626
3	220/220	1200 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	35627
4	220/220	600 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	35624
5	220/220	600 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	35628
6	220/220	600 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	
7	220/220	300 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	35618
8	220/220	300 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	
9	220/220	300 / -	1	-40	≤-40	≤-40	Flowmeter	Duration Test	
10	220/250	60 / -	1	-40	≤-40	≤-40	Flowmeter	MR Survey	35647
11	220/190	60 / -	1	-40	≤-40	≤-40	Flowmeter	MR Survey	35648
12	190 / 220	60 / -	1	-40	≤-40	≤-40	Flowmeter	MR Survey	35649
13	220/220	0.1/0.1	20	-40	≤-40	≤-40	Bellows	Pulse Test	35644
14	220/220	0.2/0.2	20	-40	≤-40	≤-40	Bellows	Pulse Test	35645
15	220/220	0.5/0.5	20	-40	≤-40	≤-40	Bellows	Pulse Test	35646

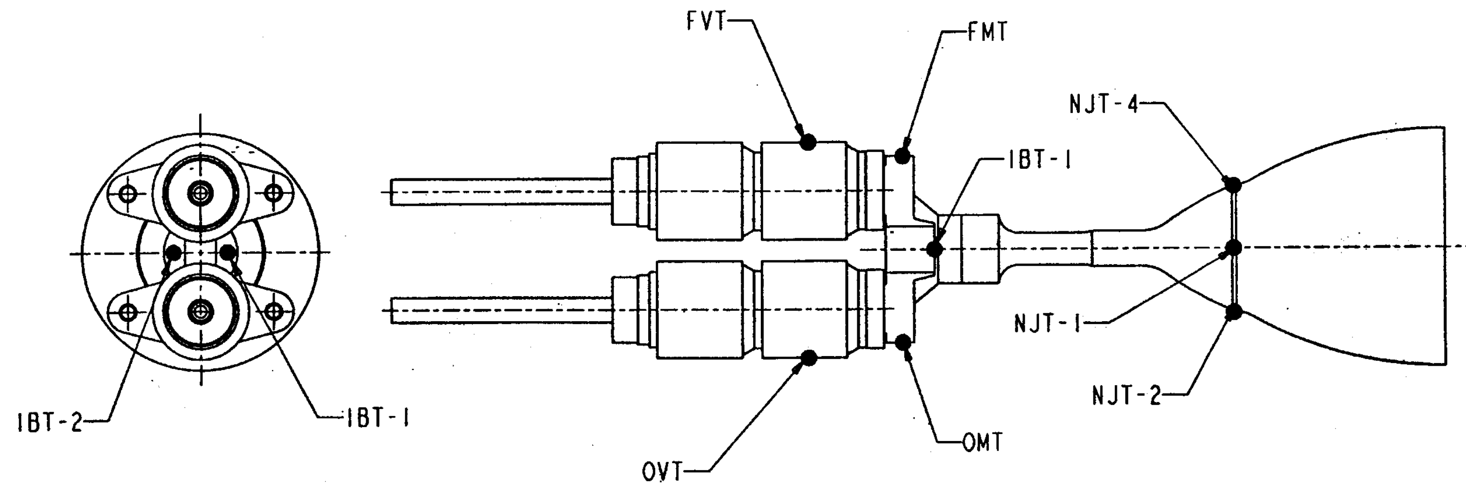


Figure 14. Thermocouple Locations

6.0 TEST RESULTS

Table 5 provides a summary of all the tests conducted during this program. The detailed test data is given in Volume 2. Most of the tests were of 30s or 60s duration and for these tests, the results shown in Table 5 are at the end of the run. For the long duration 600s and 1200s runs, a data slice at 60s and at the end of the run are given in Table 5. Given in Table 5 are results for thrust, Isp, mixture ratio(MR), propellant flowrates(total, fuel, oxidizer), propellant temperatures and valve and injector temperatures before the start of the test. Also shown is a value for Pc which is calculated from the measured thrust, assuming a value for the thrust coefficient(Cf) of 1.770. C* is calculated from this estimated value for Pc. The maximum chamber temperature measured during the test using the Agema infrared camera is also given in Table 5.

Test 35607 was a 30s test which was terminated at 28s when an excessive amount of fuel was observed coming from the nozzle. This test was successfully rerun as Test 35610. It was suspected that there was gas in the oxidizer line during Test 35607 which caused poor combustion. This thruster has a high fraction of the fuel in the barrier so the chamber runs relatively cold, however, a consequence of this is that it is difficult to burn all the fuel, particularly when the feed pressure is low and the propellant is cold. Typically, at the low feed pressure conditions fuel was observed collecting around and vaporizing from the rim of the nozzle.

Test 35464 was a pulse test with a 0.500/0.500s duty cycle. This test was terminated before completion when fuel was observed coming from the nozzle and the strip chart data showed no evidence that combustion was occurring in the chamber. The previous two pulse tests with duty cycle of 0.100/0.100s and 0.200/0.200s ran successfully.

Test 35649 was terminated early since the MON-25 oxidizer was exhausted early in the test. During the test, fuel was observed running out of the nozzle. The test data shows the mixture ratio decreasing during the test, indicating the oxidizer was being exhausted.

No.	Test No.	Date	Feed Pres. OFP/FFP bar	Run Time On/Off sec	Pulses	Data Slice sec	Thrust N	Isp sec	Pc bar	Cstar m/s	MR	Propellant Flowrates Total gm/s	Fuel gm/s	Ox gm/s	Max Cham Temp C	Injector Temp C	Propellant Temperatures Ox C	Fuel C	Pre-test Hardware T Ox Valve C	Fuel Valve C	Injector C	COMMENTS
1	35606	28 Oct	15.2/15.2	60	1	60	10.15	277.6	7.12	1504	1.616	3.726	1.424	2.301	note 4	51	25	25	22	22	23	
2	35607	"	6.9/6.9	28	1																	Test terminated
3	35608	29 Oct	13.8/13.8	60	1	60	8.77	254.8	6.17	1384	1.585	3.507	1.357	2.150		48	21	21	25	23	26	
4	35609	"	10.3/10.3	60	1	60	7.03	249.8	4.94	1355	1.546	2.869	1.127	1.742		48	21	22	25	24	34	
5	35610	"	6.9/6.9	60	1	60	4.54	216.7	3.18	1171	1.254	2.135	0.947	1.188		48	21	21	24	23	31	
6	35611	"	17.2/17.2	60	1	60	11.04	280.5	7.77	1522	1.604	4.010	1.540	2.470		49	21	22	21	19	28	
7	35612	"	20.7/20.7	60	1	60	12.68	285.1	8.93	1549	1.624	4.534	1.728	2.806		50	21	21	19	19	31	
8	35613	"	15.2/17.2	60	1	60	10.28	266.1	7.21	1440	1.801	3.938	1.406	2.532		50	21	21	24	24	30	
9	35614	"	15.2/13.1	60	1	60	9.12	264.6	6.41	1435	1.396	3.514	1.467	2.048		46	21	21	22	20	33	
10	35615	"	13.1/15.2	60	1	60	8.77	244.0	6.17	1324	1.837	3.662	1.291	2.371		48	21	21	19	17	29	
11	35616	"	15.2/15.2	.10/.10	20	na	0.85	201.1			1.622	0.433	0.165	0.268		26	18	19	20	19	22	
12	35617	"	15.2/15.2	60	1	60	10.59	288.3	7.43	1561	1.600	3.745	1.440	2.304		47	20	21	16	16	18	Test cell pressure = 0.20 psia
13	35618	4 Nov	15.2/15.2	300	1	60	9.48	264.9	6.67	1438	1.865	3.647	1.273	2.374	623	19	-38	-39	-38	-43	-40	
	"	"	"	"		300	9.43	268.0	6.65	1457	1.794	3.588	1.284	2.304		18	-39	-39				
14	35619	"	6.9/6.9	60	1	60	3.20	130.7	2.26		1.798	2.499	0.893	1.606		3	-38	-39	-39	-43	-41	
15	35620	5 Nov	10.3/10.3	60	1	60	6.72	231.4	4.74	1259	1.920	2.960	1.014	1.946	293	11	-40	-41	-43	-51	-38	
16	35621	"	15.2/15.2	60	1	60	10.28	288.4	7.23	1564	1.900	3.633	1.253	2.380	359	17	-41	-41	-53	-58	-52	Test cell pressure = 0.20 psia
17	35622	"	17.2/17.2	60	1	60	10.24	265.5	7.20	1441	1.931	3.929	1.341	2.589	591	17	-41	-41	-53	-60	-43	
18	35623	10 Nov	20.7/20.7	60	1	60	11.97	278.5	8.43	1514	1.858	4.381	1.533	2.848	888	23	-40	-41	-51	-46	-40	
19	35624	"	15.2/15.2	600	1	60	8.05	225.0	5.65	1217	1.963	3.649	1.232	2.417	662		-41	-41	-62	-51	-29	
	"	"	"	"		600	9.57	268.6	6.73	1458	1.840	3.631	1.278	2.352		18	-39	-39				
20	35625	11 Nov	15.2/15.2	1200	1	60	9.52	266.7	6.68	1444	1.882	3.640	1.263	2.377	673		-40	-41	-43	-46	-41	
	"	"	"	"		1200	9.61	269.5	6.77	1463	1.852	3.636	1.275	2.361		20	-41	-39				
21	35626	"	15.2/15.2	1200	1	60	8.90	249.8	6.27	1357	1.948	3.632	1.232	2.400	678		-40	-42	-47	-49	-43	
	"	"	"	"		1200	9.66	269.8	6.79	1464	1.848	3.648	1.281	2.367		19	-41	-39				
22	35627	"	15.2/15.2	1200	1	60	9.35	266.6	6.57	1446	1.942	3.573	1.214	2.359	893		-42	-42	-44	-46	-39	
	"	"	"	"		1200	9.52	266.2	6.69	1442	1.801	3.647	1.302	2.345		18	-39	-38				
23	35628	"	15.2/15.2	600	1	60	9.70	273.0	6.82	1481	1.865	3.622	1.264	2.358	679		-37	-39	-44	-46	-51	
	"	"	"	"		600	9.61	271.8	6.76	1474	1.802	3.605	1.286	2.318		18	-40	-38				
24	35629	12 Nov	6.9/6.9	30	1	30	4.72	197.2	3.31	1067	1.663	2.438	0.916	1.523		26			-3	-5	-2	
25	35630	"	10.3/10.3	30	1	30	6.94	235.8	4.90	1283	1.664	3.001	1.126	1.874	262	28	-2	-2	-4	-7	-8	
26	35631	"	15.2/15.2	60	1	60	9.83	268.7	6.91	1456	1.650	3.731	1.408	2.323	629	32	-2	-3	-9	-12	-1	
27	35632	"	17.2/17.2	30	1	30	10.90	275.2	7.66	1492	1.644	4.038	1.527	2.511	712	32	-1	-2	-11	-12	-4	
28	35633	"	20.7/20.7	30	1	30	12.42	280.1	8.74	1522	1.644	4.518	1.709	2.809	878	34	-1	-1	-10	-12	-3	
29	35634	"	6.9/6.9	30	1	30	4.98	212.6	3.51	1155	1.787	2.390	0.857	1.532		17	-17	-17	-22	-23	-22	
30	35635	"	10.3/10.3	30	1	30	7.30	245.9	5.12	1330	1.806	3.025	1.078	1.947	268	19	-18	-19	-23	-24	-22	
31	35636	"	15.2/15.2	60	1	60	10.01	272.9	7.04	1480	1.764	3.740	1.353	2.387	679	24	-19	-19	-26	-28	-19	
32	35637	"	17.2/17.2	30	1	30	10.90	278.4	7.66	1509	1.690	3.992	1.484	2.508	732	24	-18	-19	-22	-25	-21	
33	35638	"	20.7/20.7	30	1	30	12.50	283.2	8.81	1539	1.713	4.501	1.659	2.842	912	26	-18	-19	-30	-32	-21	
34	35639	"	6.9/6.9	30	1	30	4.76	201.9	3.35	1096	1.839	2.404	0.847	1.557		11	-29	-29	-29	-32	-33	
35	35640	"	10.3/10.3	30	1	30	7.12	247.4	5.01	1344	1.793	2.934	1.050	1.883	264	14	-29	-30	-31	-33	-30	
36	35641	"	15.2/15.2	60	1	60	9.88	272.6	6.94	1477	1.782	3.694	1.328	2.366	665	19	-31	-31	-35	-38	-29	
37	35642	"	17.2/17.2	30	1	30	10.86	277.3	7.64	1505	1.787	3.991	1.432	2.559	734	21	-29	-30	-34	-36	-31	
38	35643	"	20.7/20.7	30	1	30	12.28	281.7	8.63	1528	1.801	4.444	1.587	2.858	891	23	-30	-31	-37	-38	-28	
39	35644	"	15.2/15.2	.10/.10	20	na	0.81	208.4			1.915	0.397	0.136	0.261		-34	-38	-39	-43	-44	-43	
40	35645	"	15.2/15.2	.20/.20	20	na	1.62	224.1			2.159	0.737	0.233	0.503		-22	-39	-40	-48	-49	-43	
41	35646	"	15.2/15.2	.50/.50	11	na										-42						Terminated, no combustion
42	35647	"	15.2/17.2	60	1		9.83	258.3	6.93	1404	2.127	3.881	1.241	2.640	615	18	-39	-40	-45	-47	-42	
43	35648	"	15.2/13.1	60	1		8.94	265.9	6.28	1441	1.712	3.429	1.264	2.165	482	15	-40	-41	-46	-47	-41	
44	35649	"	13.1/15.2	10	1											-29						Stopped: ox depleted during ru

TEST NOTES

1. Pc calculated from thrust assuming Cf = 1.77
2. Throat area for C* calculation is:
3. For pulse runs, values in thrust colum 0.07865 cm2
4. The chamber temperature data for Tests 35606-35617 has been lost due to an instrumentation error.

Table 5. Mars Flyer Program Test Data Summary

Figure 15 shows the time history for the propellant, fuel valve, oxidizer valve and injector temperatures for Test 35625, which was a 1200s test at the nominal 15.2 bar (220 psia) feed pressure and with –40C propellant. Figure 15 shows that the injector and valve temperatures were below –40C at the start of the run. Pre-test hardware temperatures for each test are given in Table 5. The results in Figure 15 show the valve and injector temperatures rise smoothly during the run with steady-state valve temperatures being reached after 500s and the injector temperature after about 300s.

Figure 16 shows the time history of thrust, specific impulse and mixture ratio for Test 35625. Thrust and Isp are relatively constant except for two bumps which appear to be correlated with the dips in the propellant temperatures shown in Figure 12. Figure 17 shows a higher resolution chart of the mixture ratio time history. As the injector temperature increases, the mixture ratio shifts downward reaching a steady-state value of about 1.84. The nominal mixture ratio for the thruster with 21C propellants is 1.65; the changes in propellant viscosity caused by operation with –40C propellants cause the mixture ratio to shift from a nominal 1.65 to 1.84.

Figure 18 shows the time history of the propellant flowrates for Test 35625. The flows are quite steady throughout the run. The oxidizer flow shows two small oscillations associated with the pump cycling off and being restarted. The fuel flow shows a small increase during the start transient as the injector temperature increases as this is responsible for the decrease in mixture ratio during the start transient.

Figure 19 shows the effect of propellant temperature on thruster mixture ratio for tests at the nominal 15.2 bar (220 psia) feed pressure. The data covers the propellant temperature range of 25C to –42C (77F to –44F). The data trend shows that as propellant temperature decreases, the mixture ratio increases. As discussed in Section 2, this is caused by the large increase in the fuel viscosity as the propellant temperature decreases. This thruster was designed for a nominal mixture ratio of 1.65 with 21C (70F) propellants and as the results show, the mixture ratio shift is

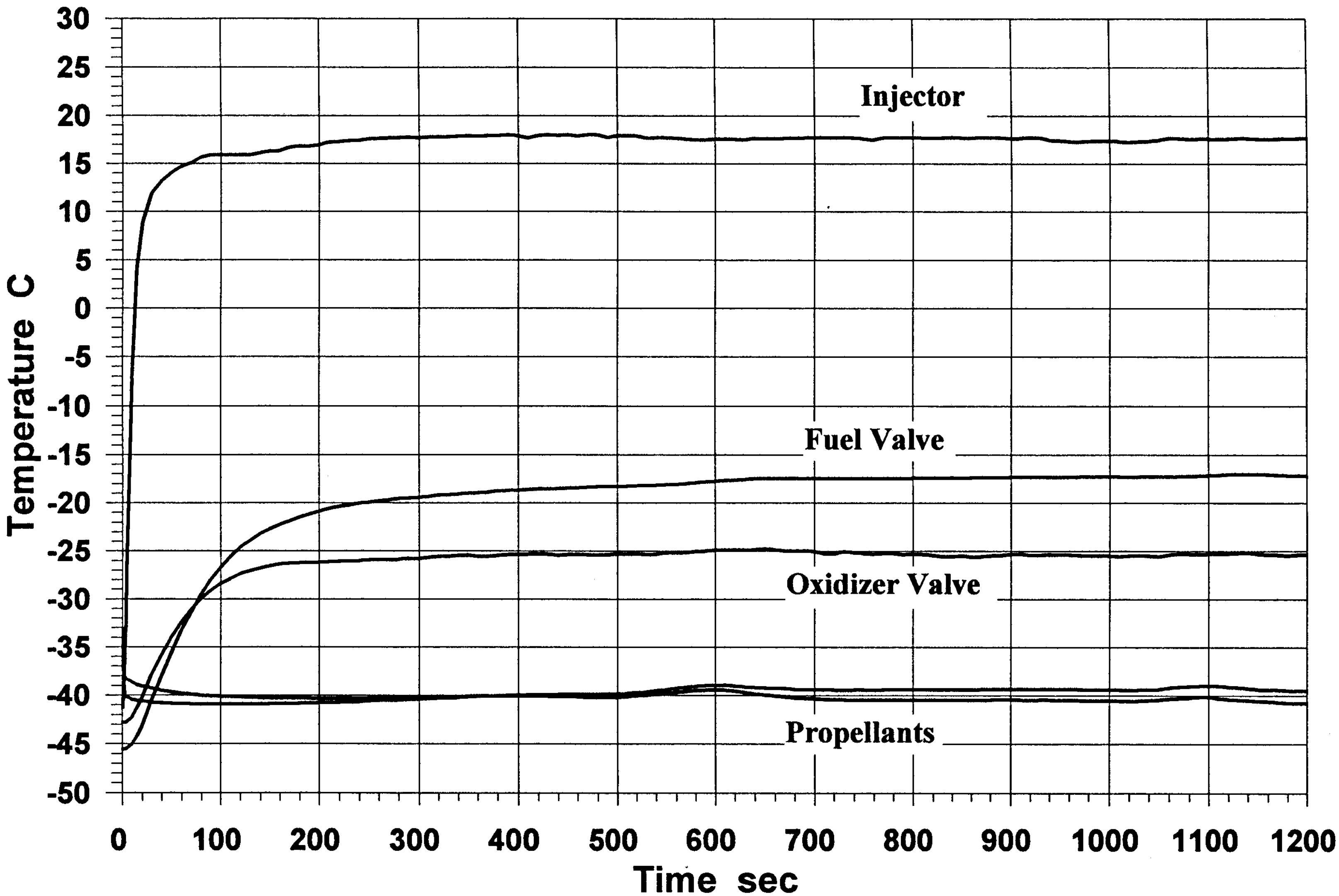


Figure 15. Temperature Time History, Test 35625

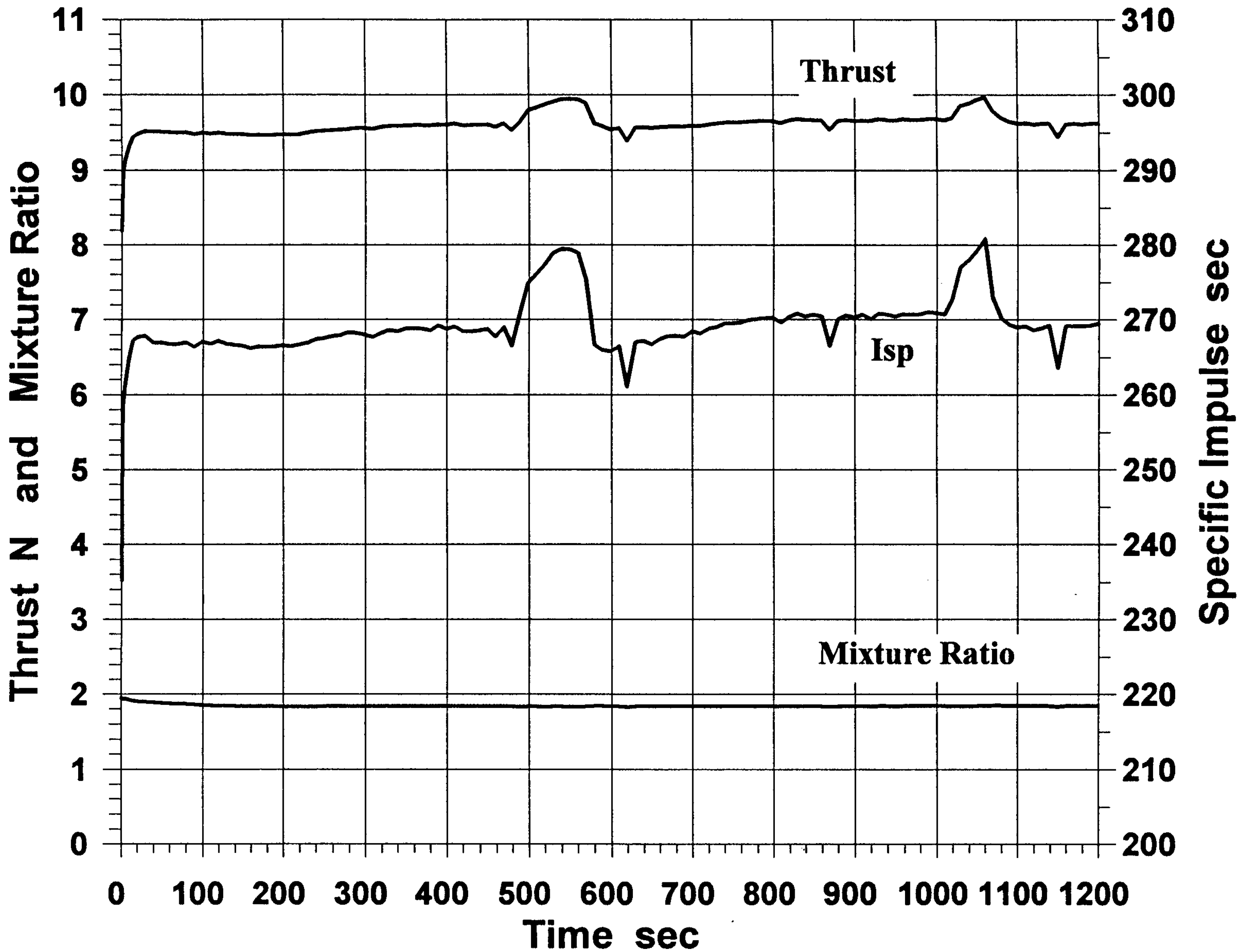


Figure 16. Thrust, Isp and MR History, Test 35625

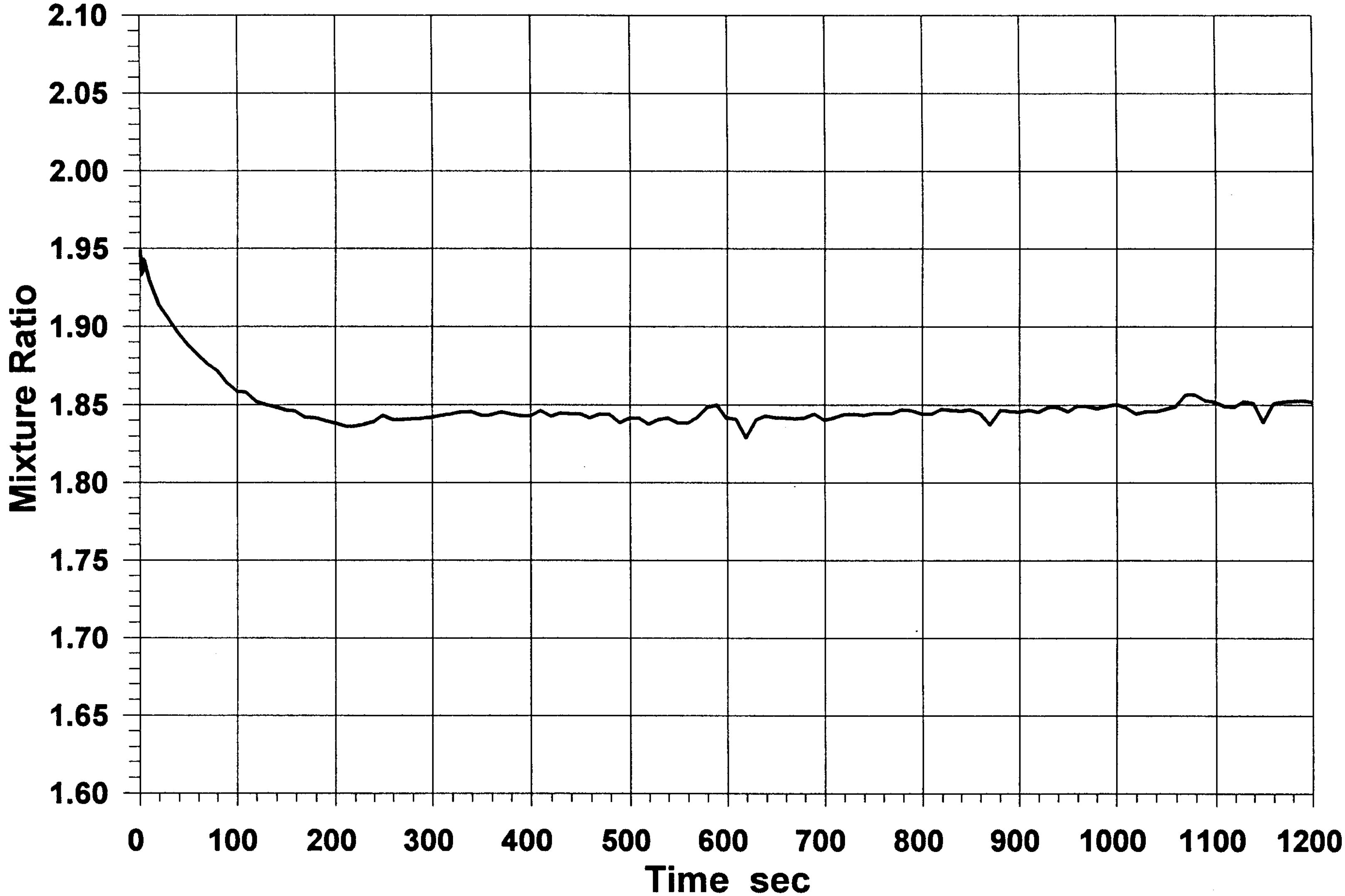


Figure 17. Mixture Ratio Time History, Test 35625

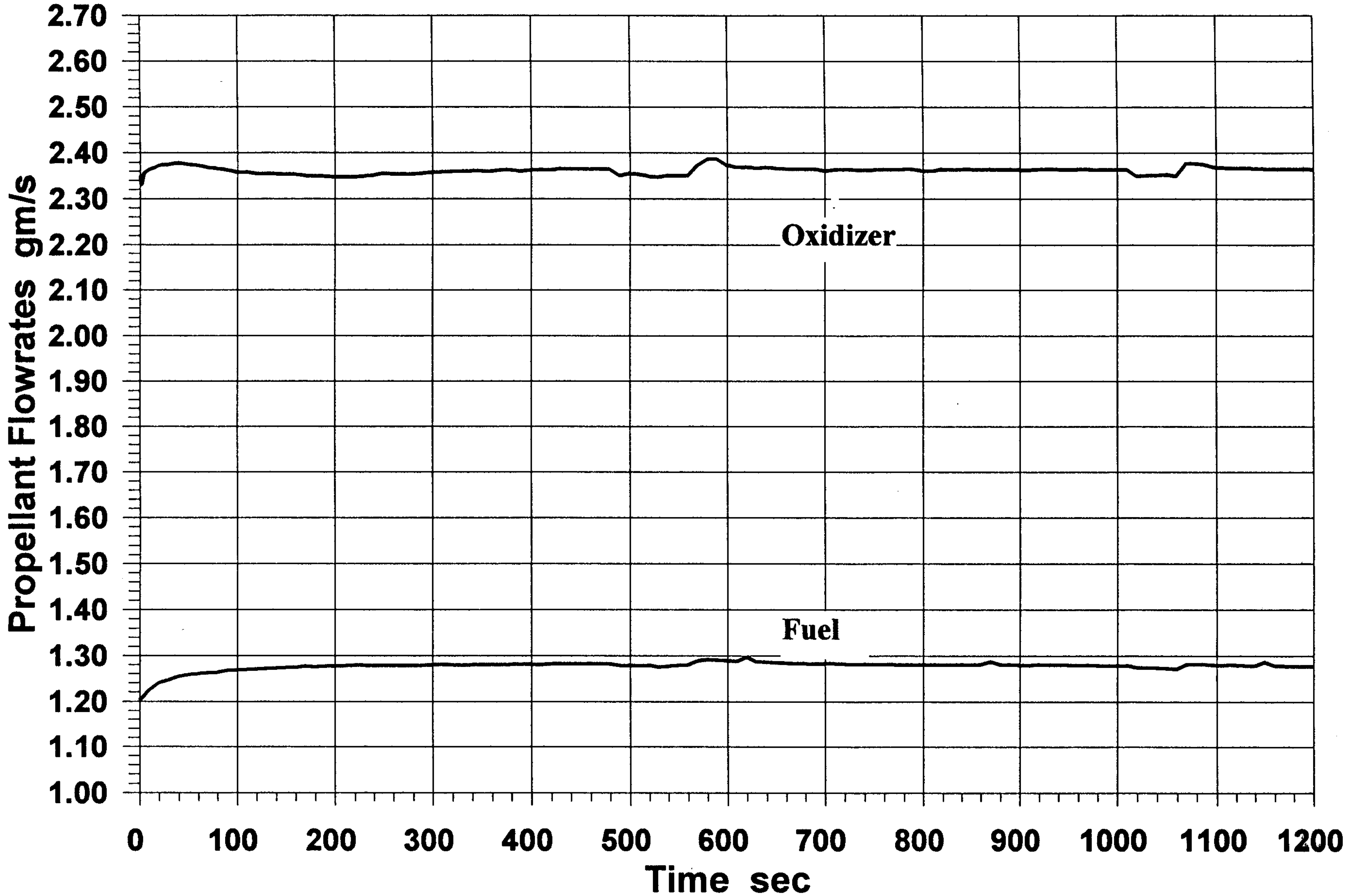


Figure 18. Propellant Flowrate Time History, Test 35625

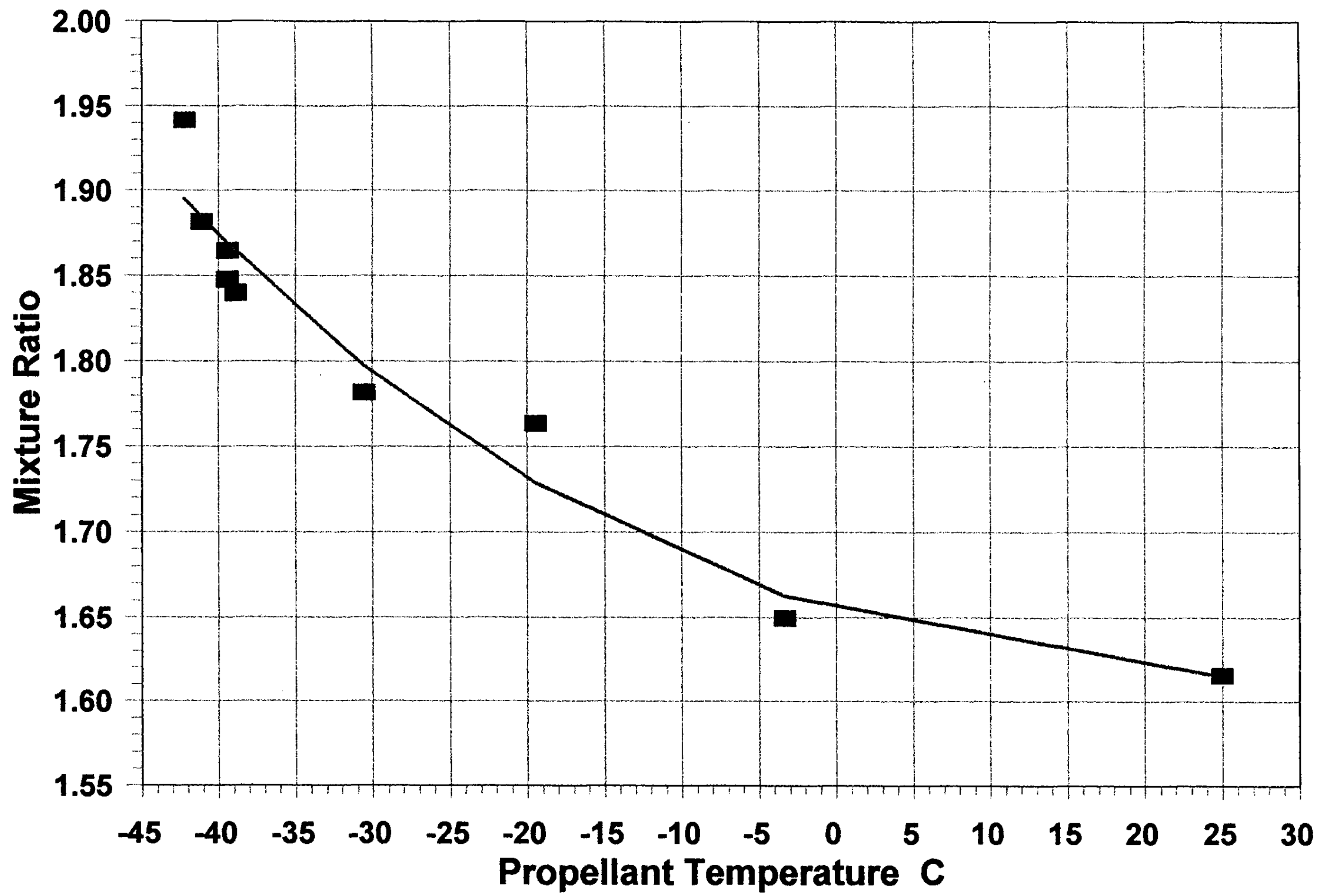


Figure 19. Thruster Mixture Ratio Variation With Propellant Temperature

modest for temperatures above about –4C (25F), but increases rapidly for temperature below this level. It should be noted that it would be relatively easy to redesign the thruster by modifying the orifices and flow passages to provide a mixture ratio of 1.65 with –40C (-40F) propellants such that optimum performance could be obtained for the Mars environment.

Figure 20 shows how thruster specific impulse is affected by propellant temperature. The effect is quite modest with Isp decreasing by about 10s as the propellant temperature is decreased from 25C (77F) to –42C (-44F). Figure 4 shows how Isp varies with MR for 21C (10F) propellants and these results show that as MR increases from 1.65 to 1.85 for the 16 bar tests, Isp decreases by about 5s. This suggests that the decrease in Isp due to operation with cold propellants is caused about equally by the induced shift in mixture ratio and by the propellant temperature, itself.

Figure 21 shows the effect of both feed pressure and propellant temperature on the thruster mixture ratio. The results show that at all propellant temperatures, except the 21C (70F) case, mixture ratio is relatively insensitive to feed pressure. With 21C (70F) propellants, the mixture ratio decreases as feed pressure decreases and falls sharply for feed pressures below about 10.3 bar (150 psia). This occurs because at low feed pressures, the oxidizer tends to boil in the injector causing a decrease in the oxidizer flowrate and a downward shift in mixture ratio. This does not occur with cold propellants since they can absorb the injector heating without reaching the saturation temperature of the oxidizer.

Figure 22 shows how thruster specific impulse varies with feed pressure and propellant temperatures. The results show that Isp decreases as feed pressure and propellant temperatures decrease. The effect of propellant temperature has been discussed. The decrease with feed pressure is typical for a thruster of this type and is due to a lower combustion efficiency as mass flowrate and chamber pressure decrease.

ARC's 2 lbf thruster uses a high level of fuel barrier cooling to maintain low chamber and injector temperatures when the thruster is firing. The chamber temperatures are measured

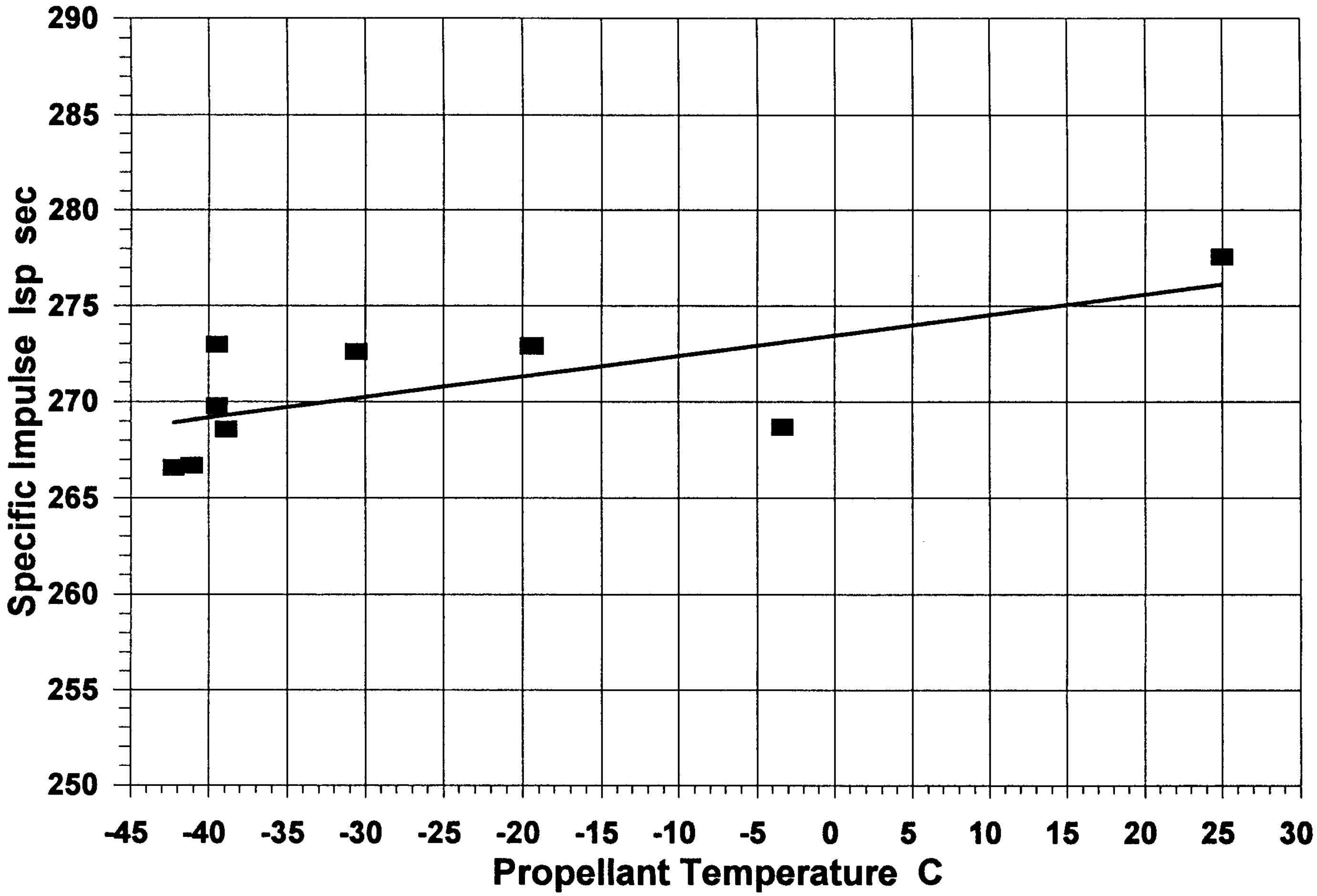


Figure 20. Specific Impulse Variation With Propellant Temperature

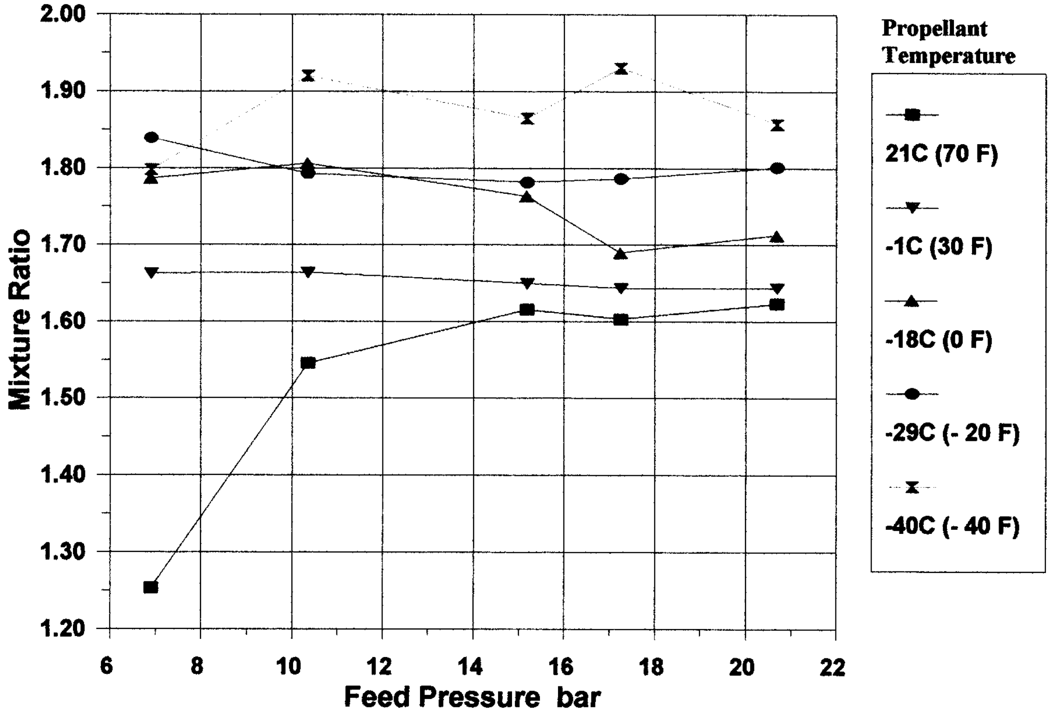


Figure 21. Effect of Propellant Feed Pressure and Temperature on Mixture Ratio

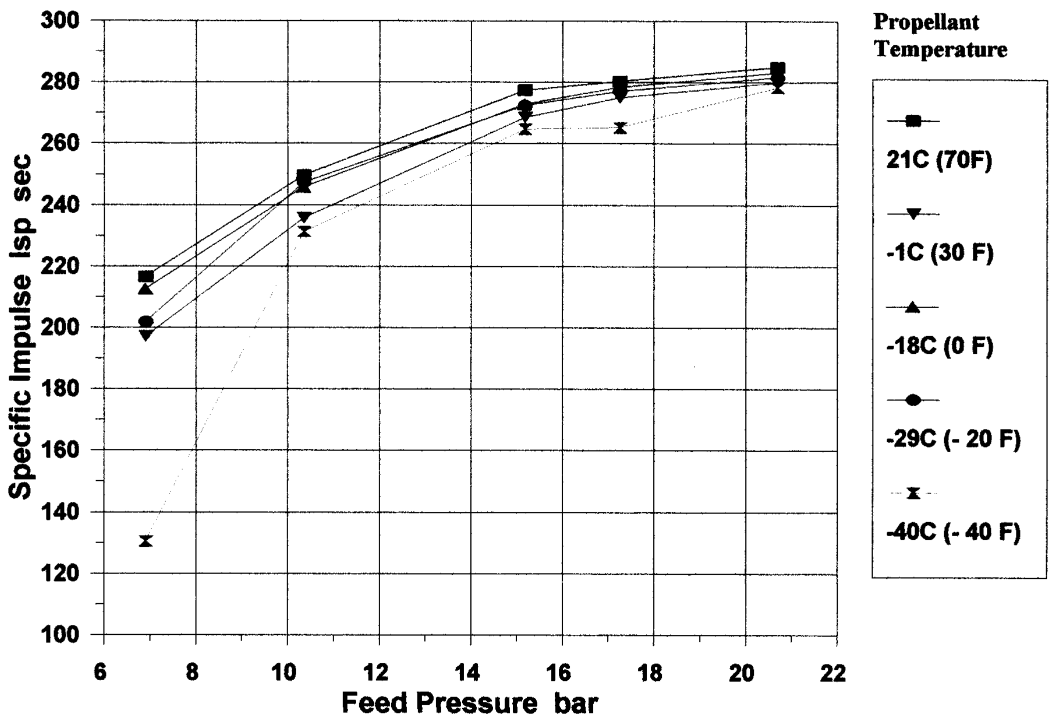


Figure 22. Effect of Propellant Feed Pressure and Temperature On Specific Impulse

using an Agema infrared camera and the data in Table 5 that chamber temperatures are below about 870C (1700F) at all conditions. At the lower feed pressures, chamber temperatures were below the infrared scanner measurement threshold of 260C (500F).

Figure 23 shows how the injector temperatures are affected by the propellant feed pressure and temperature. Due to the high level of barrier cooling in this thruster, the injector temperatures are quite low. At the normal operating condition with 21C (70F) propellant, the injector temperature is about 49C (120F) over the entire range of feed pressures. As shown in Figure 23, the injector temperature decreases as the propellant temperature is reduced. As propellant temperature is reduced, the injector temperatures show a greater sensitivity to feed pressure. The strong influence of propellant temperature on the injector temperature is expected since the propellant provides the primary cooling for the injector.

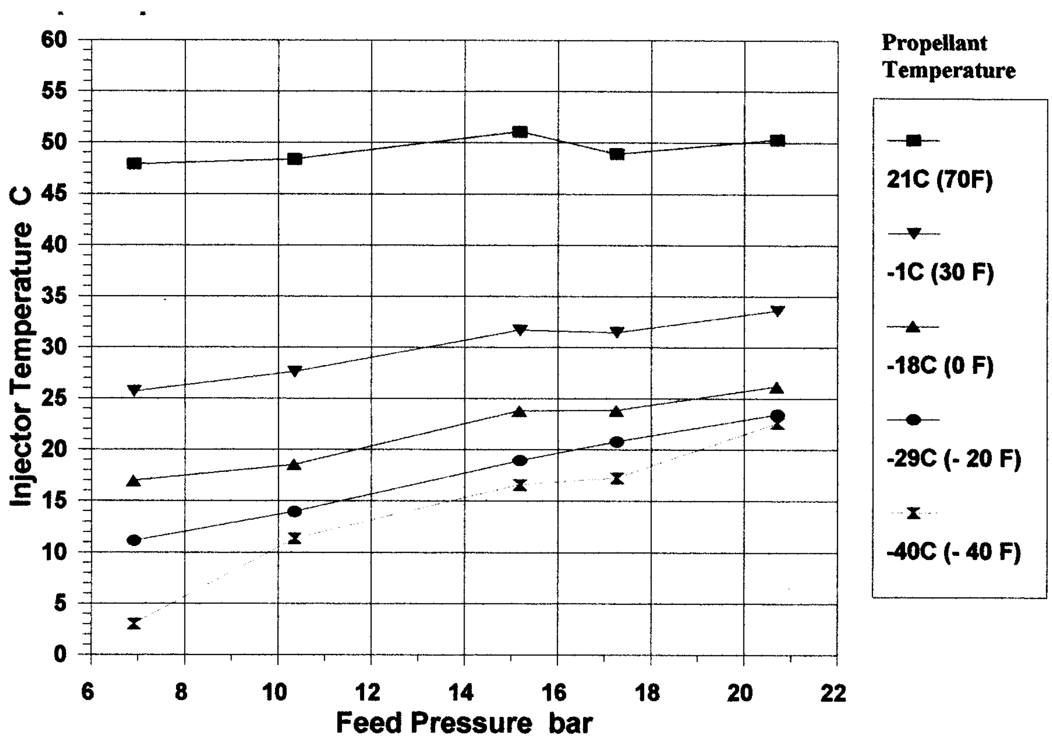


Figure 23. Effect of Propellant Feed Pressure and Temperature On Injector Temperature

7.0 CONCLUSIONS

ARC's 10N thruster was successfully tested with propellant cooled to –40C and the thruster and valves cooled below this level. The thruster was operated successfully over the entire propellant temperature range of 21C to –40C (70F to –40F) and feed pressure range of 6.9 – 20.7 bar (100-300 psia) required by the NASA-approved test matrix. While the thruster was not designed to operate at these conditions, it did demonstrate the capability to operate successfully at the temperature conditions expected to be encountered in the Mars environment.

When operating with –40C propellants, the thruster experienced a shift in mixture ratio from the nominal value of 1.65 with 21C propellants to about 1.85. This shift is caused by the increase in the MMH viscosity as propellant temperatures are reduced with a consequent reduction in the MMH flowrate and increase in mixture ratio. The increase in mixture ratio and the lower energy content of the –40C propellants cause a slight reduction of specific impulse of about 10s compared to the baseline value with 21C propellants.

During this program, ARC successfully demonstrated the ability to manufacture MON-25 as required to conduct these tests. The procedures are now in place to make this propellant for any future activities. Further, ARC developed and demonstrated propellant conditioning systems for the fuel and oxidizer with the capability to deliver propellants to the thruster at –40C with a tolerance of ±1C for tests of any duration. This ability to make propellants and to precisely control propellant temperatures provides NASA the demonstrated capability to conduct such Mars environment tests in the future.

The testing conducted on this program was exploratory in nature. The purpose was to use an existing thruster design to determine if there were any unexpected problems which would be encountered when operating a thruster with MON-25/MMH propellants at –40C. The conclusion to be derived from these tests is that no such problems were uncovered and that a properly designed thruster should have no difficulty operating in the Mars environment.

8.0 RECOMMENDATIONS

Propellant Properties

There is a lack of data on the properties of MON-25, particularly at the low temperature conditions of interest for the Mars environment. Data on the thermodynamic and transport properties of this propellant over the 100C to –50C range is needed to support thruster design and performance evaluation activities. It is recommended that NASA compile a database of MON-25 properties, identify deficiencies in the database, and initiate a program to obtain the required data.

Thruster Design

The results in this report show that a thruster designed for 21C propellants will operate somewhat differently when tested with –40C propellants. The reasons for the mixture ratio shift and specific impulse decrease observed herein are understood. The mixture ratio shift can be corrected by either modifying the injection orifices in the thruster or possibly, by using a different set of trim orifices upstream of the valve. Modifying the design will allow one to recover the Isp lost to the mixture ratio shift. It is recommended that in any future activity to examine propulsion for the Mars environment, NASA provide a set of design objectives for the thruster and consider modifying an existing design to meet these requirements before initiating a test evaluation.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.

1. AGENCY USE ONLY (*Leave blank*)	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	April 2001	Final Contractor Report

4. TITLE AND SUBTITLE

Mars Flyer Rocket Propulsion Risk Assessment
ARC Testing

5. FUNDING NUMBERS

WU–755–B4–01–00
NAS3–99197

6. AUTHOR(S)

Atlantic Research Corporation

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Atlantic Research Corporation
6686 Walmore Road
Niagara Falls, New York 14304–1697

8. PERFORMING ORGANIZATION REPORT NUMBER

E–12642

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, DC 20546–0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

NASA CR—2001-210709

11. SUPPLEMENTARY NOTES

Project Managers, Brian Reed, 216–977–7489, and James Biaglow, 216–977–7480, Power and On-Board Propulsion Technology Division, NASA Glenn Research Center, organization code 5430.

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified - Unlimited
Subject Category: 20 Distribution: Nonstandard

Available electronically at http://gltrs.grc.nasa.gov/GLTRS

This publication is available from the NASA Center for AeroSpace Information, 301–621–0390.

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

This report describes the investigation of a 10-N, bipropellant thruster, operating at –40 °C, with monomethylhydrazine (MMH) and 25% nitric oxide in nitrogen tetroxide (MON-25). The thruster testing was conducted as part of a risk reduction activity for the Mars Flyer, a proposed mission to fly a miniature airplane in the Martian atmosphere. Testing was conducted using an existing thruster, designed for MMH and MON-3 propellants. MON-25 oxidizer was successfully manufactured from MON-3 by the addition of nitric oxide. The thruster was operated successfully over a range of propellant temperatures (–40 to 21 °C) and feed pressures (6.9 to 20.7 kPa). The thruster hardware was always equal or lower than the propellant temperature. Most tests were 30- and 60-second durations, with 600- and 1200-second duration and pulse testing also conducted. When operating at –40 °C, the mixture ratio of the thruster shifted from the nominal value of 1.65 to about 1.85, probably caused by an increase in MMH viscosity, with a corresponding reduction in MMH flowrate. Specific impulse at –40 °C (at nominal feed pressures) was 267 sec, while performance was 277 sec at 21 °C. This difference in performance was due, in part, to the mixture ratio shift.

14. SUBJECT TERMS

Liquid rocket propellants; Rocket testing; Mixed oxides of nitrogen; Mars flyer

15. NUMBER OF PAGES 54

16. PRICE CODE A04

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500

Standard Form 298 (Rev. 2-89)
Prescribed by ANSI Std. Z39-18
298-102