(NASA-CR-157956) EVALUATION CF THE SSRCS ENGINE WITH HYDRAZINE AS A FUEL, PHASE 2 Final Report (Marquardt Corp.) 57 p HC A04/MF A01 CSCL 21I

N79-13181

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JET PROPULSION LABORATORY

CALIFORNIA INSTITUTE OF TECHNOLOGY

PASADENA, CALIFORNIA

EVALUATION OF THE SSRCS ENGINE
WITH HYDRAZINE AS A FUEL

PHASE II

FINAL REPORT

Contract No. 954468

OCTOBER 1978

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ABSTRACT

Test firing of the Space Shuttle Reaction Control Thruster (SSRCT) was conducted to determine the characteristic velocity and chamber throat temperatures when the fuel is changed from monomethylhydrazine (MMH) to hydrazine.

Thruster performance with hydrazine was essentially as predicted. Characteristic velocity at a mixture ratio of 1.4 (equal volumetric flow) for hydrazine was 5180 feet/second compared with 5110 feet/second at a mixture of 1.6 for MMH. Specific impulse with a 22:1 nozzle is calculated to be 280 lbf-sec/lbm.

Thermal performance, as measured by chamber throat temperature, was dramatically different (colder) than predicted. Throat temperatures of 2330°F were predicted from a cooling model which assumed a reactive liquid cooling film. Throat temperatures of 800°-1000°F were measured. An attempt to reconcile the differences between film cooling predictions and the measurements indicated that an evaporation (nonreactive fluid) model rather than a decomposition (reactive fluid) model best fits the test data at O/F = 1.4. At higher mixture ratios, a transition to temperatures typical of the reactive fluid model were observed.

This suggests that criteria must be developed to define the applicability of each model.

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I. INTRODUCTION

The Space Shuttle Reaction Control Thruster (SSRCS) is a pulse-modulated engine which provides 870 pounds of force for attitude control and delta velocity capability for the Space Shuttle Orbiter. It uses monomethylhydrazine (MMH) and nitrogen tetroxide (N_2O_4) as propellants at a nominal mixture ratio (O/F) of 1.6 and is capable of providing minimum pulsewidths of 40 milliseconds.

A program was initiated to experimentally measure the rocket performance and thermal behavior when using hydrazine instead of MMH as the fuel. Phase I of the program provided analytical predictions of the effects of switching from MMH to hydrazine upon rocket performance, thermal behavior, stability and pulsing operation. Reference 1 presents the results of that analysis.

This report completes Phase II of the program. The objective of the Phase II program is to obtain experimental data on the performance of the Space Shuttle Reaction Control Thruster when hydrazine and nitrogen tetroxide are used as the propellants. A comparison is made with the Phase I predictions.

The test results which are presented in this report are for the Marquardt SSRCS P/N X30850 (Thruster 2-4). This thruster had been tested previously, both at The Marquardt Company (Reference 2) and at NASA/JSC (Reference 3), with MMH/N_2O_4. These data were used for comparing the performance of the same thruster with hydrazine. The internal configuration of the injector, chamber and nozzle are identical to the present SSRCS Primary Thruster. However, Thruster 2-4 has a bolted exit nozzle bell rather than the welded bell used on the SSRCS.

II. THRUSTER TEST HARDWARE

The Thruster 2-4 injector is the first of the SSRCS injectors to have the standardized injector configuration. It is designed to provide a nominal 870 pounds of thrust at an inlet pressure of 238 psia (dynamic) in a vacuum environment. Figure 1 shows the SSRCS Primary Thruster overall configuration. Figure 2 is a photograph of the injector face. The injector P/N X20850 uses a single injection ring and 84 unlike doublets to encourage good secondary mixing by means of overlap and interactive splashing between the closely-spaced doublets. The doublet angles are selected to produce two propellant momentum angles (one inward and the other outward) so as to provide a more uniform propellant mass distribution within the chamber.

The nominal injector doublet and film cooling holes are listed below.

	Inner	Outer
Oxidizer, Main	0.0315	0.0315
Fuel, Main	0.0225	0.0225
Fuel Cooling, 20 degrees	0.020	-
Fuel Cooling, 45 degrees	-	0.0165

The film cooling flow for this thruster was approximately 25% of total fuel flow and is about 1% higher than subsequent thrusters.

Thruster 2-4 was the last engine to use the bolted connection between the combustion chamber and the exit nozzle bell. The location of the bolted section is shown in Figure 3. Subsequent thrusters used a welded connection at the same location. Thruster 2-4 is a developmental engine, and the injector and chamber can be disassembled at another bolted flange interface, also shown in Figure 3. Tests of welded configurations verify that the differences in external construction do not affect the validity of the test results.

Fire testings were performed with the columbium chamber/nozzle, P/Ns X30958 and X30084. This thruster was borrowed from NASA/JSC with the provision that the testing with hydrazine be conducted in a manner which would not cause damage.

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AFT MODULE THRUSTER ENVELOPE
(-0001 ± Z)

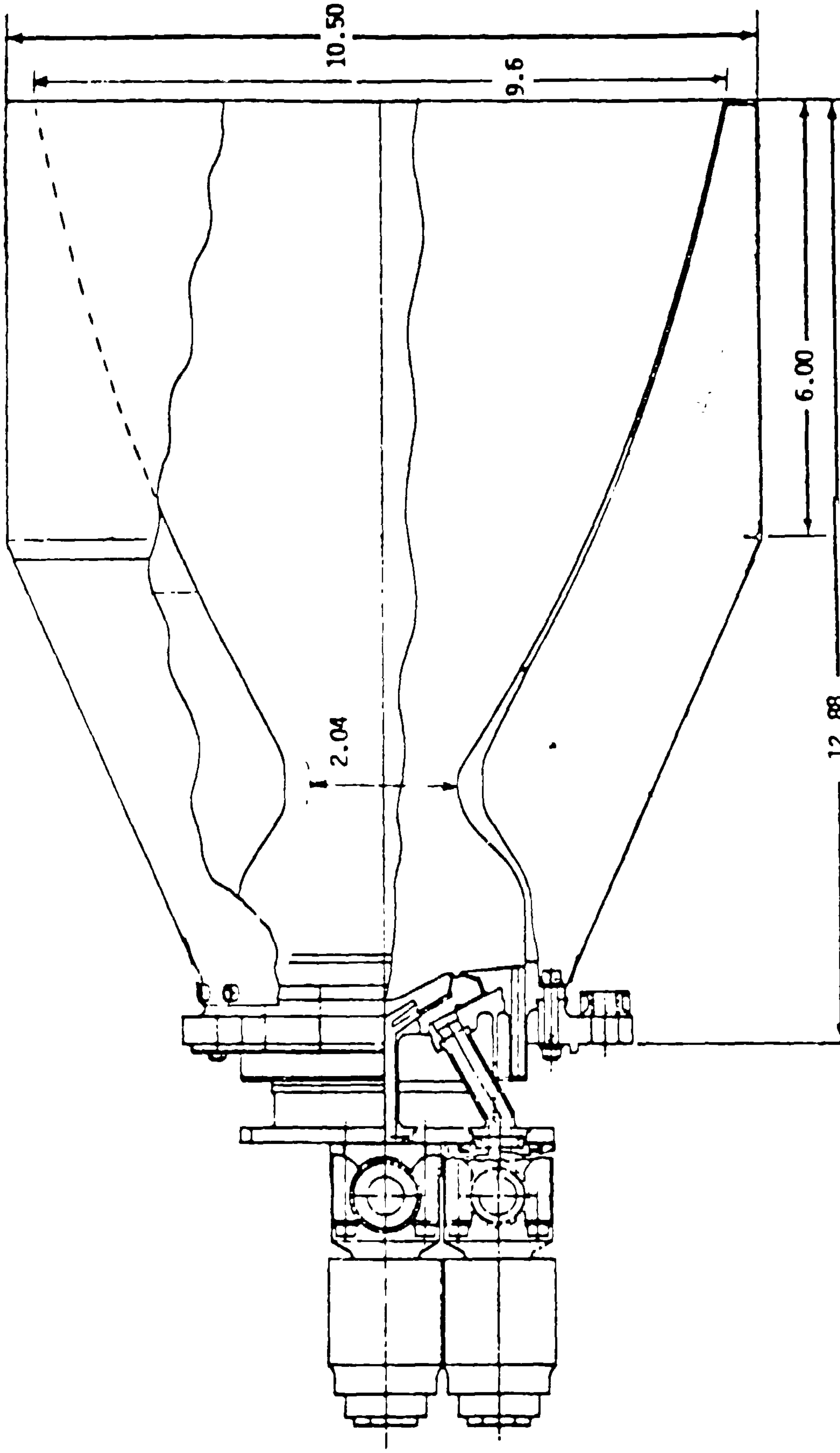
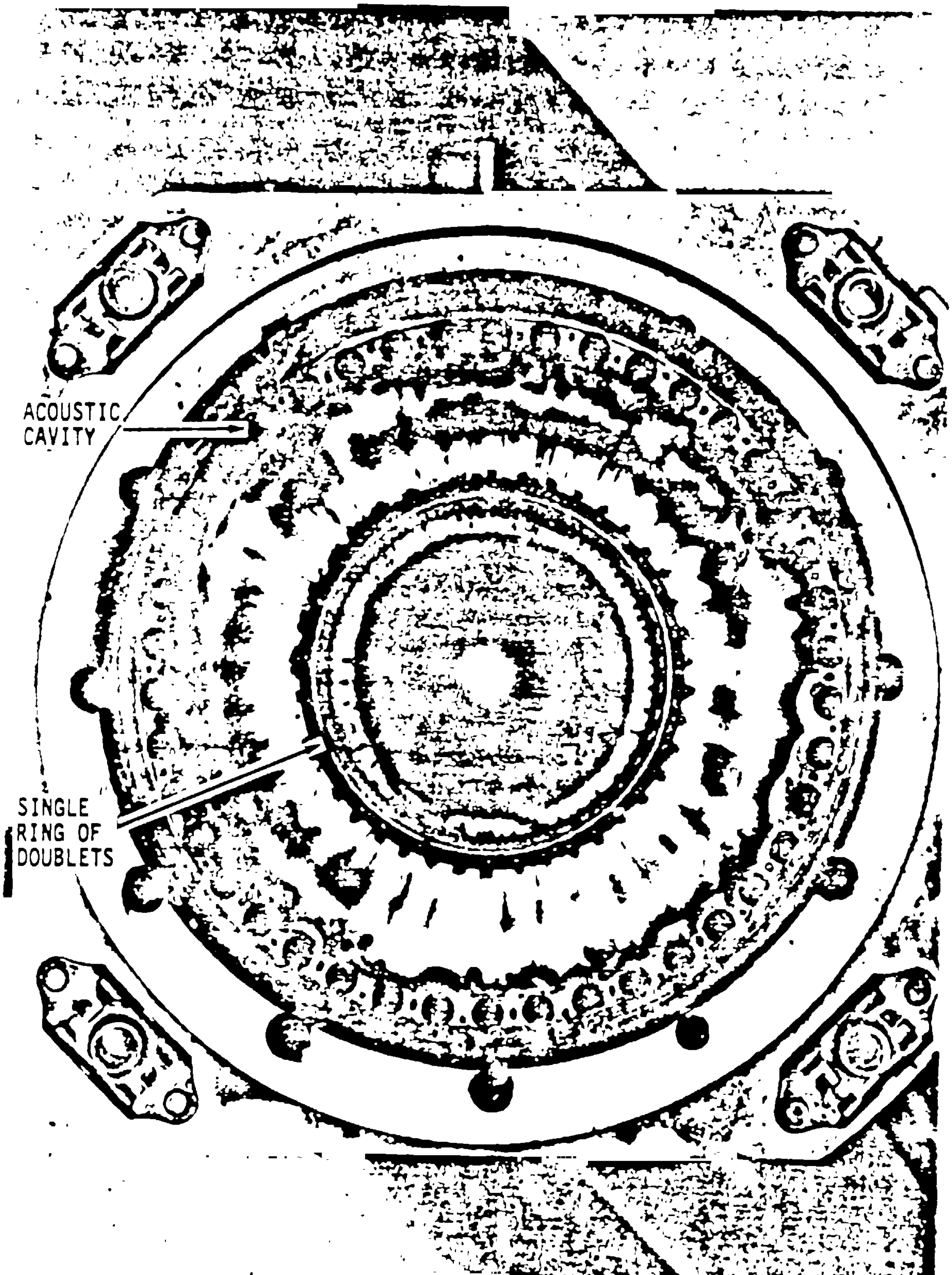


FIGURE 1



NEG. 76-352-631

2-4 INJECTOR ASSEMBLY P/N X30850 S/N 001

THRUSTER 2-4 WITH BOLTED NOZZLE BELL AND CHAMBER

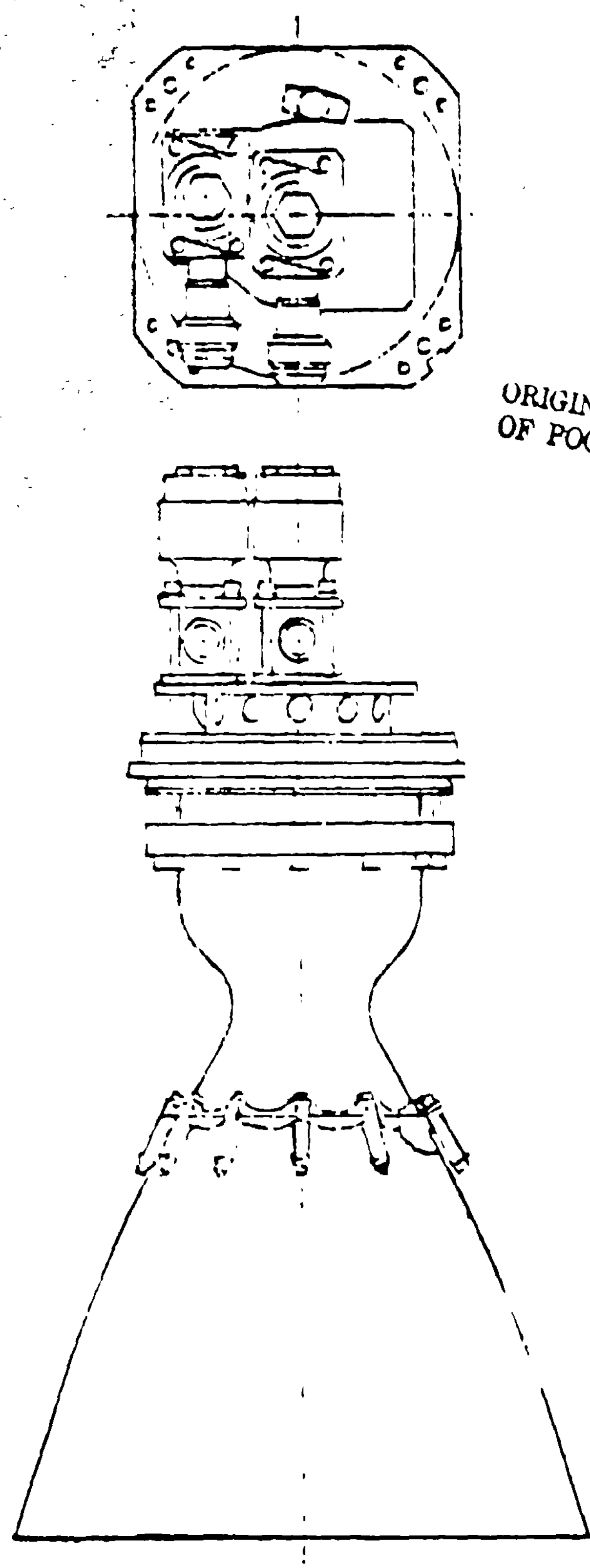


FIGURE 3

III. TEST FACILITY DESCRIPTION

All firing tests were conducted in Pad F of the Marquardt Precision Rocket facility. Marquardt Cell No. 1 was originally proposed for tests, since it was operational, fully-instrumented, and used for the previous tests of Thruster 2-4. However, due to the constant usage of Cell 1 for the testing and delivery of MMH-fueled thrusters by the high priority Space Shuttle program, Cell 1 was not available. Instead, Pad F, which was originally used for the development testing of the SSRCS thruster, was reactivated for the hydrazine test firings. Pad F is a sea level test facility and does not have a thrust stand. Rocket performance is evaluated by measuring the chamber pressure and the characteristic velocity.

Pad F consists of a vertical stand upon which the thruster is mounted with the exhaust nozzle pointed downward into a steam ejector operated exhaust scrubber. The primary function of the exhauster system is to collect and clean the exhaust gases rather than to provide a low exhaust pressure. Figure 4 shows the installation of Thruster 2-4 in Pad F. The thruster combustion chamber can be seen just below the mounting fixture. Dyna-flex insulation, by Johns-Manville (the light-colored pad shown on top of the large-diameter exhaust ducting), was used to seal the gap between the chamber exhaust nozzle and the plate covering the exhaust ducting. As finally installed, the Dyna-flex pad was held down by a close-fitting plate to prevent it from being sucked into the exhaust system.

The propellant system in Pad F consists of a hydrazine and a nitrogen tetroxide delivery system. A diagram of this system is shown in Figure 2 of the Test Plan (MTP 0254). The propellant tankage has a capacity of 27 gallons.

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THRUSTER 2-4 IN PAD F

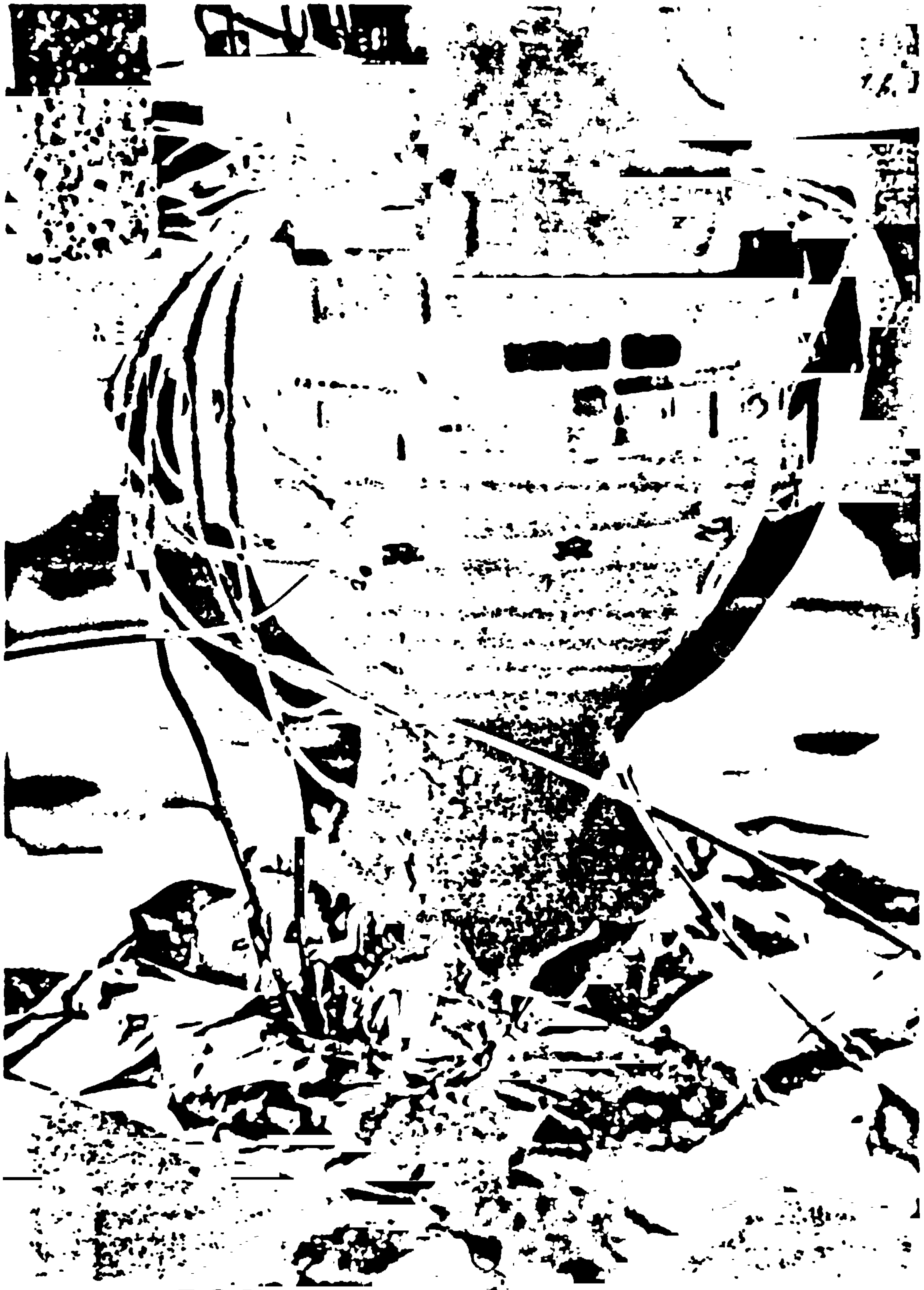

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FIGURE 4

IV. TEST PROGRAM DESCRIPTION

MTP 0254 presents the Test Plan for the fire testing of Thruster 2-4. It consisted of three test segments. In the first test series (Appendix A of MTP 0254) thruster characteristics at design thrust and mixture ratio (870 lbs force at O/F = 1.4) were obtained. Facility and instrumentation were also checked out. In the second test period, Appendices B and C were implemented. For Appendix B, characteristic velocity was measured over a wide range of mixture ratios. The third segment (Appendix C) documented the chamber temperatures under steady state operating conditions at different mixture ratios. The Test Plan describes the instrumentation, its location, and some operating procedures.

Although the primary purpose of the test firing was to determine the characteristic velocity (and, hence, specific impulse) and thermal behavior, special instrumentation was used to monitor combustion instability. In addition, a switch was installed to automatically shutdown the thruster (to prevent damage) in the event that instability was detected. The special instrumentation consisted of a high-frequency response accelerometer mounted on the injector head and a Kistler pressure transducer mounted in a special boss already built into the Thruster 2-4 chamber. The pressure transducer was added after the first series of tests (Appendix A) because the accelerometer had not functioned properly and some visual evidence suggested that resonant burning might be occurring in the chamber. The pressure transducer provides a redundant and more direct measure of resonant burning.

V. INJECTOR FLOW CHARACTERISTICS

The SSRCS dual momentum injector produces an inner and an outer combustion zone, and thruster performance is optimized by adjusting the fuel and oxidizer flows to each zone. The nominal equal volumetric flow mixture ratio (O/F) of the rocket with hydrazine is 1.4. Because fuel is used for liquid film cooling of the combustion chamber, the mixture ratios in the core combustion zones are higher than 1.4. The list below presents the flow splits and the zone mixture ratios based upon water flow tests of the injector. These agree closely with those measured previously for this engine.

N_2H_4	Inner	32.9% of total fuel flow
	Outer	42.0% of total fuel flow
	Cooling	25.1% of total fuel flow
N_2H_4	Inner	46.8% of total oxidizer flow
	Outer	53.2% of total oxidizer flow

$$\text{Inner Zone O/F Ratio} = \frac{46.8}{32.9} \times 1.4 = 1.99$$

$$\text{Outer Zone O/F Ratio} = \frac{53.2}{42.0} \times 1.4 = 1.77$$

for overall mixture ratio = 1.4

$$\frac{\text{O/F Inner}}{\text{O/F Outer}} = 1.2$$

In addition to documentation of the split between various ring flows, the circumferential distribution documentation tests are shown in Figure 5 for both the fuel and the oxidizer. Distribution characteristics appeared to be reasonably similar on both fuel and oxidizer, indicating a uniform O/F ratio on a circumferential basis. Boundary layer cooling distribution tests were made to document characteristics of each of the individual streams prior to firing. Results of these tests are tabulated in Tables I and II, for each of two tests completed. Although the percent deviations on some of the cooling holes are quite large, the overall cooling flow was high enough to avoid worry about local hot spots. Two 45-degree streams exhibited higher than normal flow. The high flow orifices were ones which had been reworked to remove broken drills during the manufacturing process.

Injector pressure drop characteristics (with water flow) were measured as a function of flow rate before test firing. Pressure drop characteristics without valves (corrected to propellant flow) are shown in Figure 6.

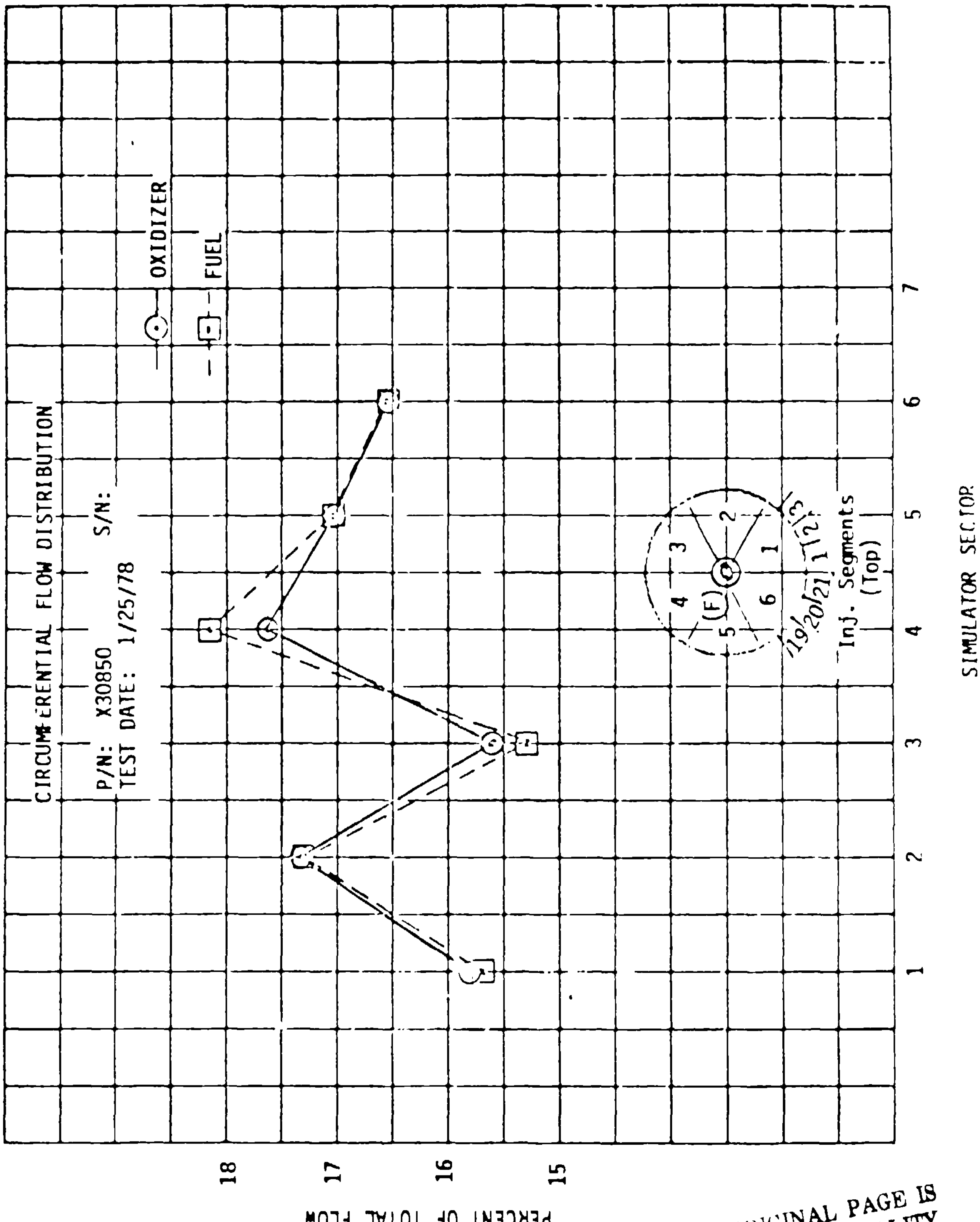
CIRCUMFERENTIAL FLOW DISTRIBUTION
P/N: X30850 S/N:
TEST DATE: 1/25/78
OXIDIZER
FUEL
Inj. Segments
(Top)
SIMULATOR SECTOR
PERCENT OF TOTAL FLOW
18
17
16
15
1
2
3
4
5
6
7

FIGURE 5

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TABLE I

SSRCT WATER FLOW TEST OF FILM COOLING RING

Average Flow Out 20-degree Orifice	5.013	ML/Sec
Average Flow Out 45-degree Orifice	4.171	ML/Sec
Percentage Flow Out 20 and 45-degree Holes	54.6	45.4

Flow Per Element

	20 Degrees		45 Degrees	
Element	Flow ML/Sec	Dev. Percent	Flow ML/Sec	Dev. Percent
1	5.16	+ 2.9	4.30	+ 3.08
2	5.37	+ 7.1	4.21	+ 0.92
3	5.28	+ 5.3	4.29	+ 2.84
4	4.15	-17.21	4.20	+ 0.68
5	5.34	+ 6.5	4.08	- 2.19
6	5.25	+10.1	3.57	-14.4
7	4.90	- 2.3	3.72	-10.8
8	5.22	+ 4.1	5.00	+19.9
9	5.16	+ 2.9	3.93	- 5.8
10	5.16	+ 2.9	4.09	- 2.0
11	5.23	+ 4.3	4.43	+ 6.2
12	4.60	- 8.2	4.17	- 0.03
13	5.06	0.9	4.30	+ 3.1
14	5.70	+13.7	4.15	- 0.51
15	4.50	-10.2	4.35	+ 4.3
16	4.50	-10.2	4.30	+ 3.1
17	4.23	-15.6	4.94	+18.4
18	4.90	- 2.3	4.05	- 2.9
19	4.94	- 1.5	3.73	-10.6
20	5.18	+ 3.3	3.72	-10.8
21	5.17	+ 3.1	4.07	- 2.4

TABLE II

SSRCT WATER FLOW TEST OF FILM COOLING RING

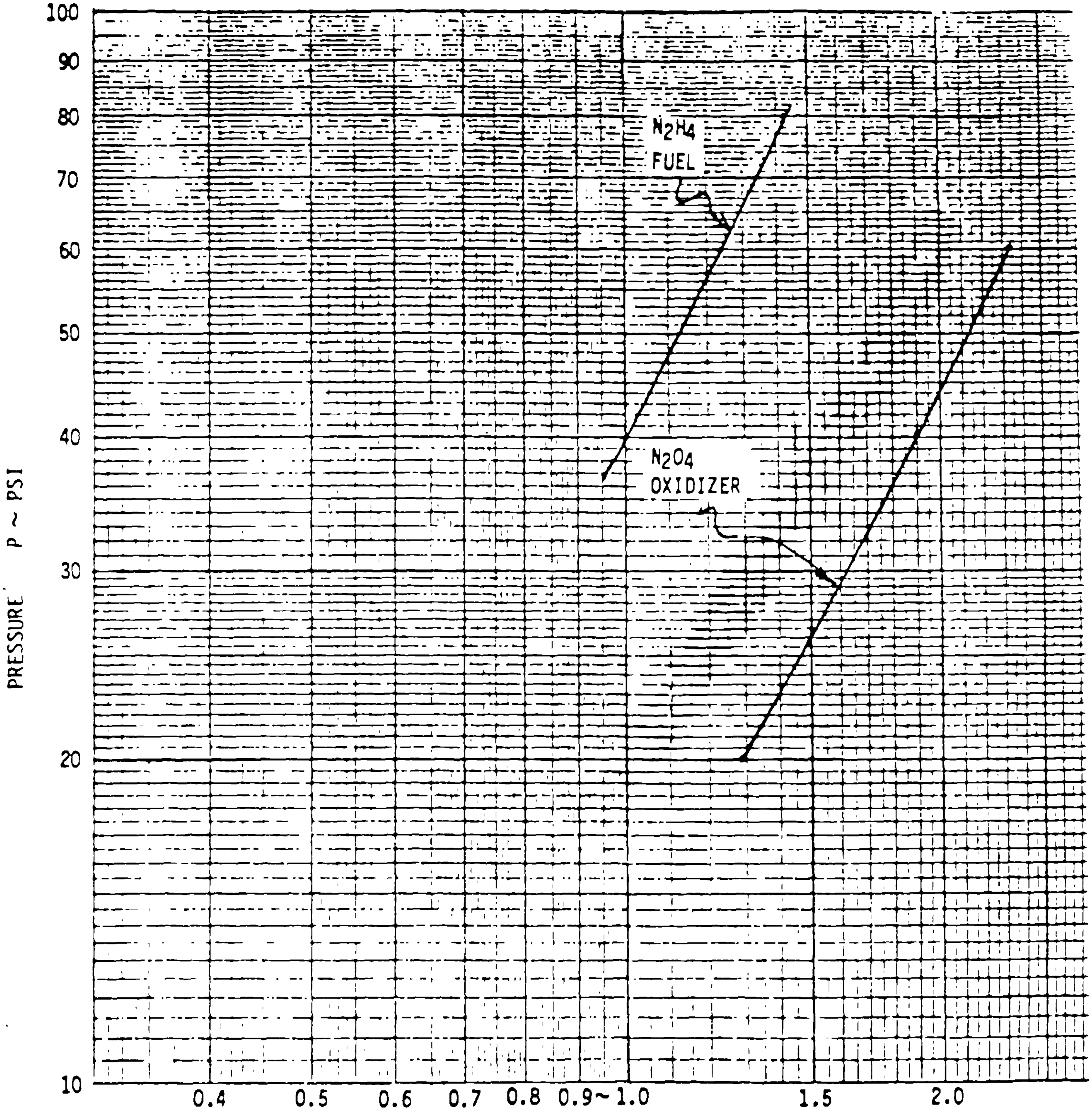
Average Flow Out 20-degree Orifice	4.977	ML/Sec
Average Flow Out 45-degree Orifice	4.211	ML/Sec
Percentage Flow Out 20 and 45-degree Holes	54.2	45.8

Flow Per Element

	20 Degrees		45 Degrees	
Element	Flow ML/Sec	Dev. Percent	Flow ML/Sec	Dev. Percent
1	5.13	+ 3.1	4.10	- 2.6
2	5.20	4.5	4.37	3.8
3	5.40	8.5	4.41	4.7
4	4.17	-15.2	4.25	0.9
5	5.32	6.9	4.13	- 1.9
6	5.65	13.5	3.70	-12.1
7	4.76	- 4.4	3.88	- 7.9
8	5.08	2.1	5.26	24.9
9	5.15	3.5	4.00	- [illegible].0
10	5.12	2.9	4.08	- [illegible].1
11	5.20	4.5	4.50	6.9
12	4.48	-10	4.18	- 0.7
13	4.94	- 0.8	4.30	2.1
14	5.58	12.1	4.14	- 1.7
15	4.42	-11.2	4.19	- 0.5
16	4.46	-10.4	4.25	0.9
17	4.34	-12.8	5.18	23.0
18	4.83	- 3.0	4.08	- 3.1
19	4.80	- 3.6	3.95	- 6.2
20	5.32	6.9	3.71	-11.9
21	5.17	3.9	3.77	-10.5

P/N: X30850
S/N
TEST DATE: 1/25/78

FLOW CALIBRATION
CONFIGURATION: THRUSTER 2-4



PROPELLANT FLOW RATE ~ PPS

FIGURE 6

VI. THRUSTER STEADY STATE PERFORMANCE

Thirty-six runs of duration from 2 to 30 seconds were conducted in the two days of testing. Table III compares the nominal performance at equal volumetric flow mixture ratios for Thruster 2-4 using MMH and hydrazine as fuels.

TABLE III

	MMH/N_2O_4	N_2H_4/N_2O_4
Equal Volumetric Flow Mixture Ratio	1.6	1.4
Characteristic Velocity ft/sec	5110	5180
Specific Impulse lbs force-sec/lbs Mass	277	280

The Tabulated Data presents the "Quick Look" data obtained and computed as the tests were being conducted. Final reduced data for the same runs are also tabulated.

Figure 7 presents the C* performance as a function of mixture ratio. The data obtained during these tests with hydrazine are shown by the circles and triangles. The close agreement between the data from the two test dates indicates the reproducibility of the instrumentation. The performance was essentially that predicted in the analytical studies. Previous test data taken on the same thruster using MMH as the fuel (Reference 2) is shown by the dashed line for comparison.

Because the testing did not include the evaluation of pulsing operation of the thruster, facility pilot-operated propellant valves which provide a slower ignition were used instead of the faster Shuttle valves. The reason for wanting the slower ignition transient is discussed in Section VIII. Figure 8 shows the start transient from one of the runs. The chamber pressure is seen to rise rapidly (in approximately 30 milliseconds after ignition) to 150 psi chamber pressure.

THRUSTER 2-4
CHARACTERISTIC VELOCITY

THRUST: 800 > lbf < 900

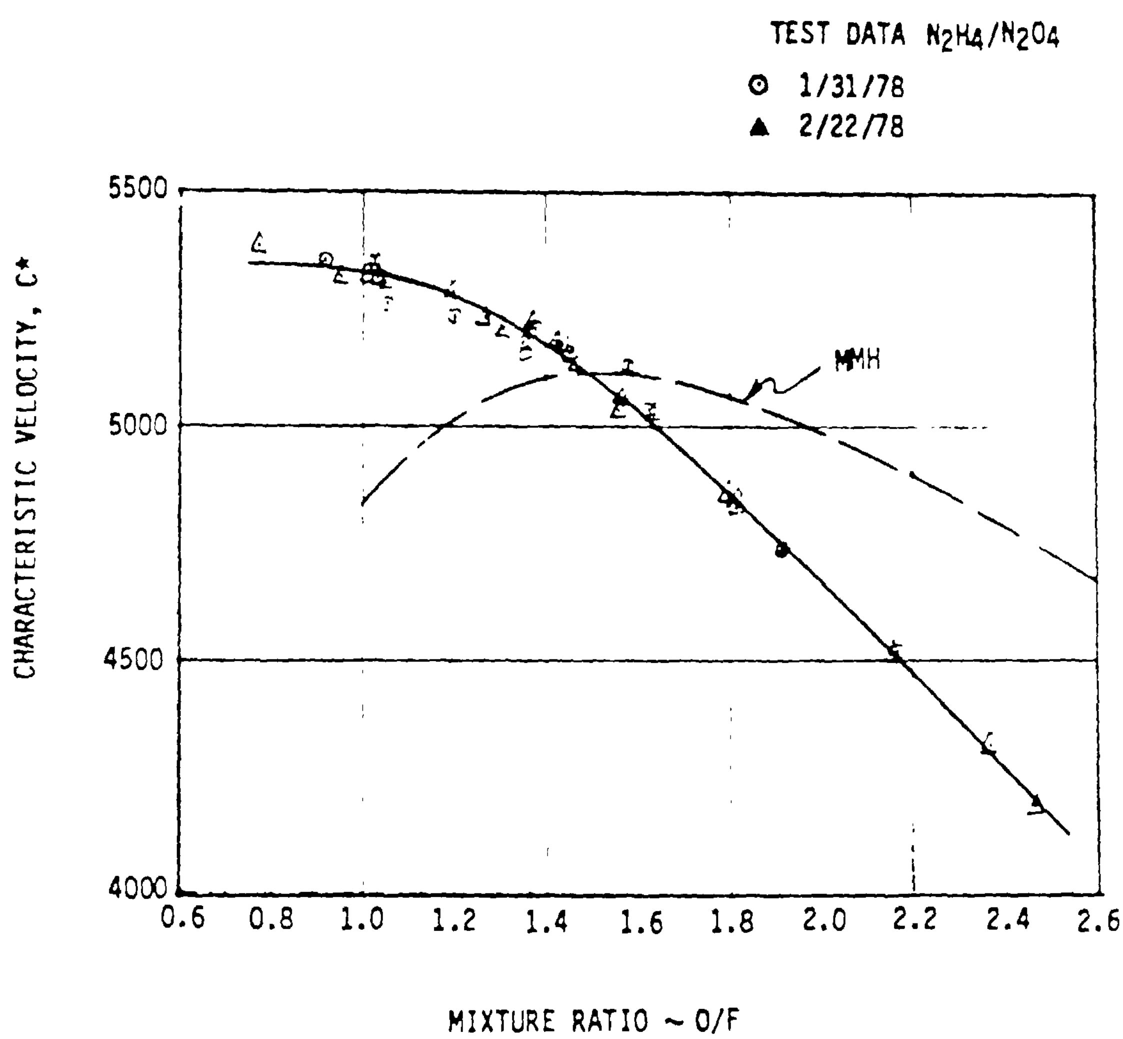


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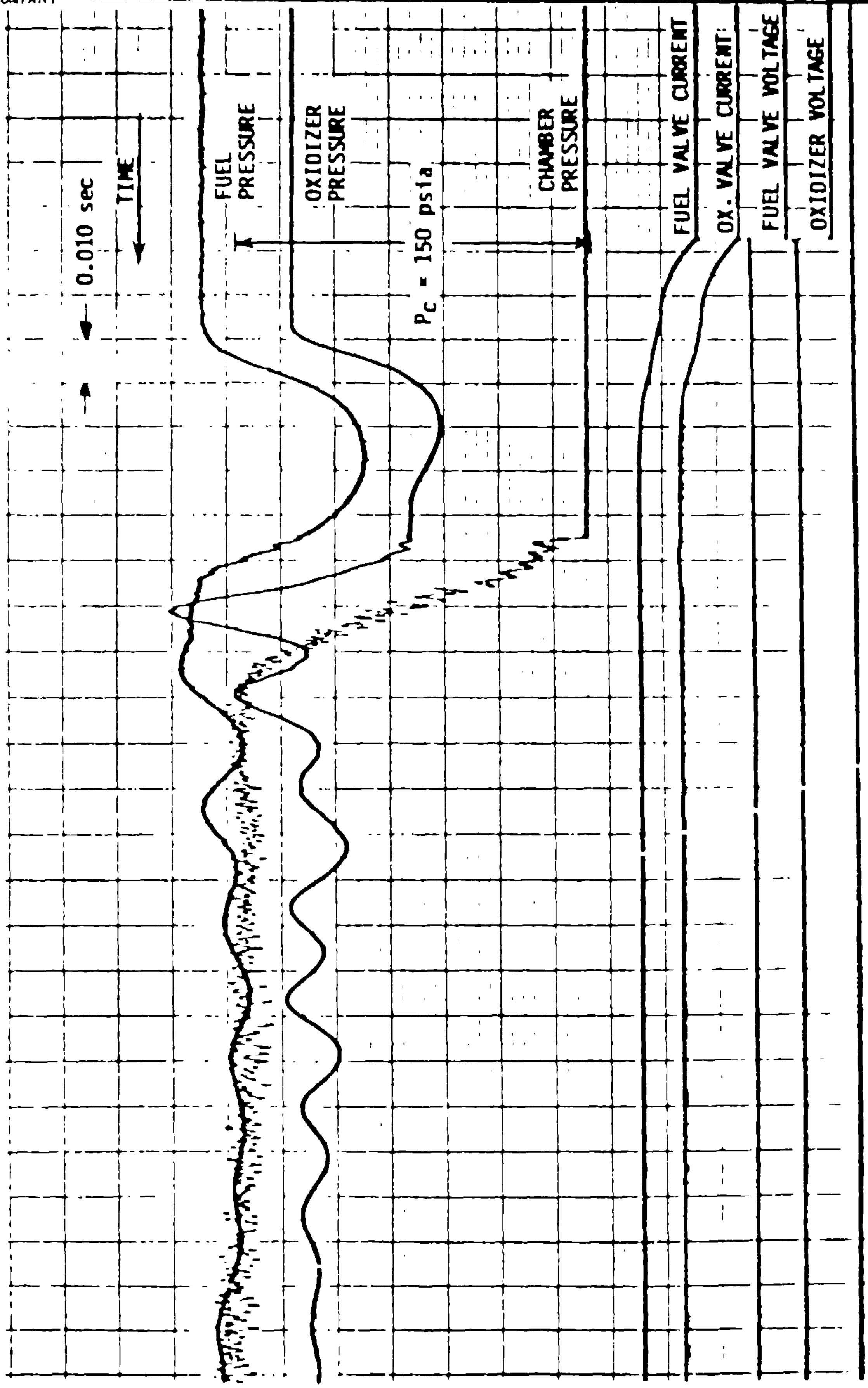
FIGURE 7



START TRANSIENT - THRUSTER 2-4

Hydrazine/N_2O_4 - Run 32 - O/F = 1.45

VII. THRUSTER THERMAL CHARACTERISTICS

A. Test Results

Thermal testing was conducted to document the thermal response of the combustion chamber. Previous testing with MMH had demonstrated that thermal equilibrium of throat temperatures is approached in about 10 to 20 seconds of operation. Maximum temperatures were observed at the exit nozzle throat or slightly downstream when MMH was used as the fuel.

A prediction that throat temperatures would be 2300°F with hydrazine was made in the Phase I study on the basis that hydrazine can be a reactive film coolant. This is approximately the temperature previously measured in tests of 100 and 300-pound force thrusters with hydrazine (i.e., 2240°-2520°F).

Figure 9 compares the steady-state throat temperatures measured during this test series with hydrazine as the fuel with those measured previously with MMH. At a mixture ratio of 1.4, average throat temperature was less than 800°F with hydrazine compared with 1350°F previously measured with the MMH as a fuel. At the nominal MMH mixture ratio of 1.6, the average throat temperature was 1550°F.

Due to air pollution restrictions in Pad F (NO_x emissions from excess N_2O_4) steady-state thermal data could not be obtained at a mixture ratio of 2.4. The throat temperatures measured after five seconds of operation are also shown on the curve to possibly provide some insight on the location of the O/F = 2.4 steady-state temperature. Further testing would be required to define the true high O/F operating temperatures and to determine if the reactive model has validity.

B. Reconciliation With Theory

Data obtained in previous testing of 100 and 300-pound force thrusters with both hydrazine and MMH as fuels have revealed that throat and chamber temperatures were 570° and 340°F greater with hydrazine than with MMH. However, in this test, hydrazine was 550°F cooler at the nominal mixture ratio. In all thrusters, the liquid fuel was used as a chamber wall film coolant and injected in a similar manner. If the liquid film is considered an inert fluid, hydrazine should provide better film cooling than MMH due to its higher heat of vaporization. However, in view of test results on prior engines where hydrazine as a film coolant caused a substantial rise in throat temperature compared with MMH, a "reactive fluid" model of the film cooling process was formulated to explain the experimental data. In Phase I of the program, it was assumed that thermal decomposition and/or bipropellant reactions could occur over the liquid film cooling layer and that these decomposition and/or bipropellant flames just above the liquid film would explain the increased evaporation rates for the more reactive hydrazine. The reactive fluid cooling model predicted a liquid length of less than one inch and a throat temperature of 2300°F. Because the reactive fluid model appeared to match previous test results obtained from smaller thrusters with very similar film cooling, it was chosen as the more realistic model in the Phase I study.

THROAT TEMPERATURES
THRUSTER 2-4
N_2H_4/N_2O_4

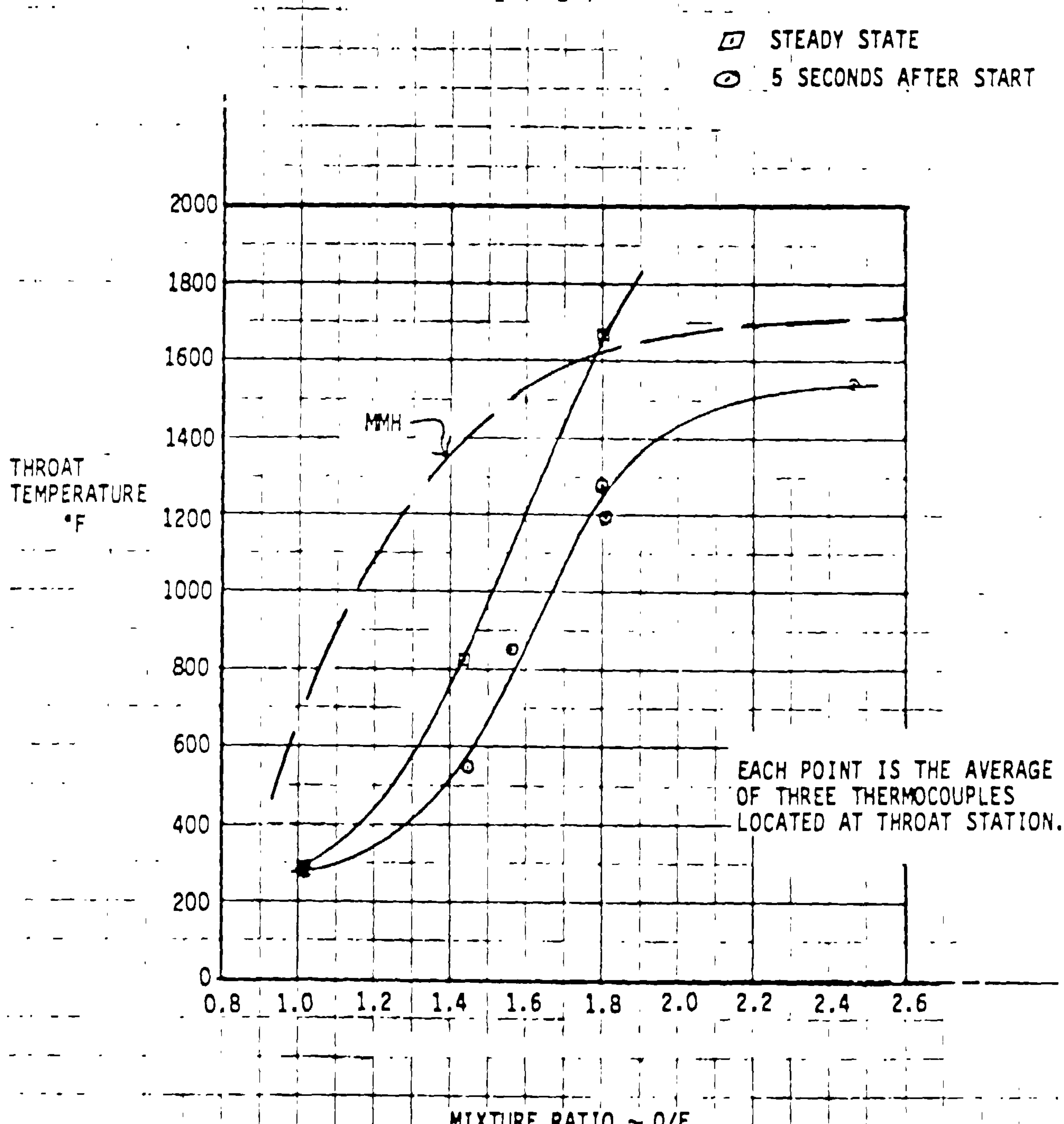


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FIGURE 9

In the Phase I report (Reference 1), the liquid film length for hydrazine was also calculated using the inert (evaporation) cooling film model. The conventional evaporation model predicted the liquid hydrazine film would extend beyond the nozzle throat (longer than 2.5 inches). This would produce nozzle throat temperatures in the neighborhood of 400°F (boiling point of hydrazine) if there was no conduction in the thick columbium nozzle throat section from the hot downstream wall. Since conduction from hotter downstream portions of the nozzle occurs, a higher throat temperature can be expected. The experimental data at O/F = 1.4 closely follows the evaporation prediction.

Figure 10 presents the axial distribution of temperature along the thruster for three mixture ratios. Figure 11 shows a color photograph of the heating pattern at the end of the O/F = 1.43 run. Both figures indicate that the inert fluid evaporation model provides a better approximation than the reactive model at a mixture ratio of 1.4, since the liquid film appears to extend beyond the throat for mixture ratios less than 1.43. However, as the mixture ratio is increased, the hydrazine throat temperatures approach and then exceed the throat temperaturs measured with MMH (Figure 8). This suggests that there is some feature of Thruster 2-4 which inhibits the decomposition or bipropellant reactions at low mixture ratios. However, a transition then occurs at higher mixture ratios where the reactive fluid model appears to be valid.

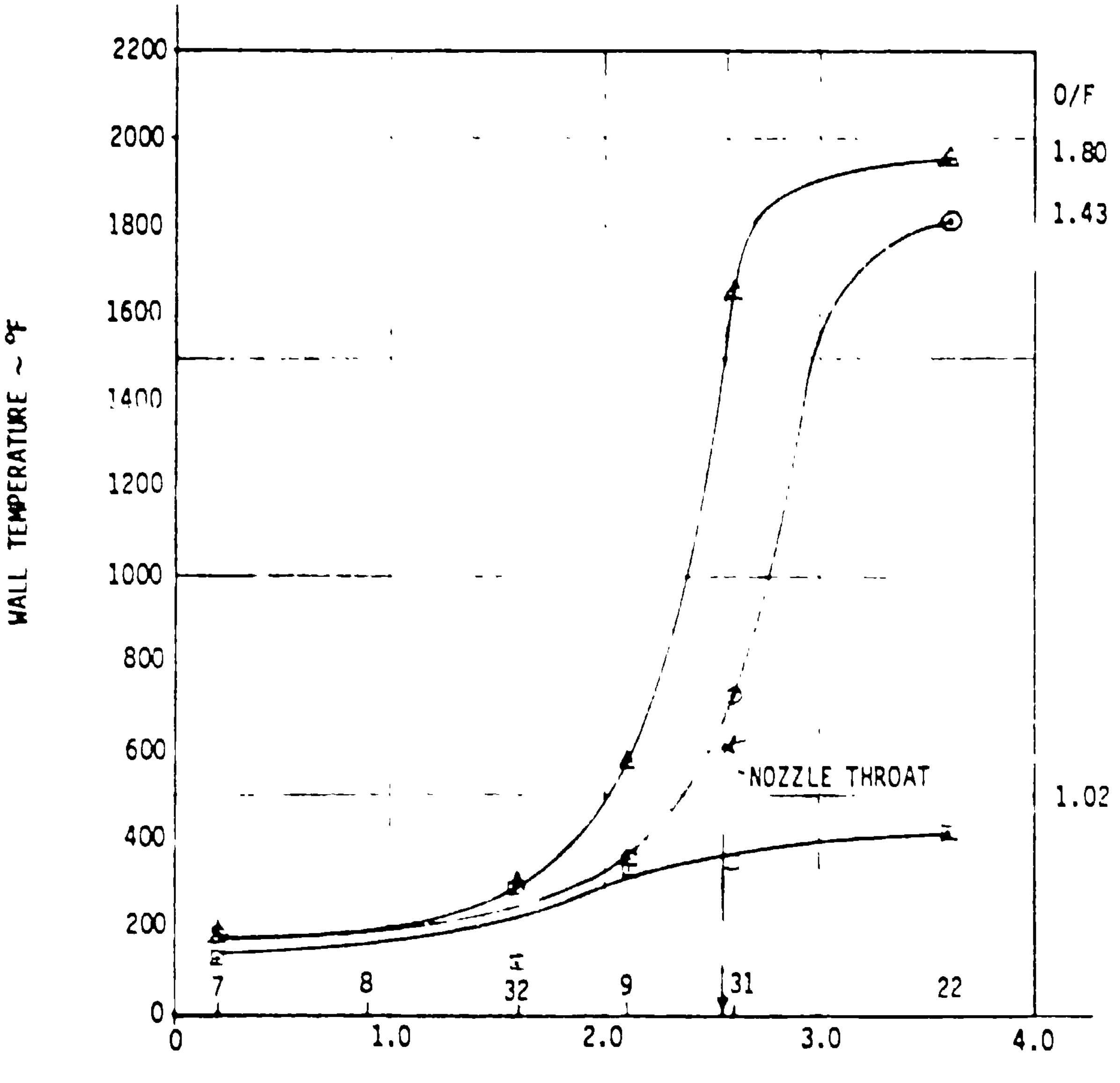
The test results, therefore, indicate that some factor in the SSRCS thruster design is preventing the occurrence of monopropellant or bipropellant reactions above the liquid film (at least over some of the mixture ratio range). One such factor could be contaminants in the hydrazine which reduce the laminar flame speed of the hydrazine monopropellant decomposition reaction. Two common contaminants in hydrazine are water and analine. Reductions of flame speeds by factors of 2 to 10 are reported for water and factors of 2 to 4 are reported for analine (References 4, 5 and 6).

A spectrographic analysis of the hydrazine fuel was made and revealed that the monopropellant grade hydrazine used in the test was not contaminated and contained less than the 1% water and 0.50% analine specified for this fuel. These concentrations of contaminants are not believed large enough to cause the observed change in liquid film length.

The experimental observations on the SSRCS thruster using hydrazine as a fuel suggest that additional study and experiments of reactive liquid films are needed to develop criteria which permit determination of when each model, the inert fluid or reactive fluid, is appropriate for the film cooling layer in rocket chambers. The internal rocket parameters which might affect this choice (i.e., chamber gas velocities, the scale or intensity of turbulence, recirculation patterns, mixture ratio, etc) have yet to be defined.

STEADY STATE
AXIAL TEMPERATURE DISTRIBUTION
THRUSTER 2-4

HYDRAZINE/N_2O_4



DISTANCE FROM INJECTOR FACE
INCHES

FIGURE 10

THRUSTER 2-4, N_2H_4/N_2O_4, O/F = 1.43

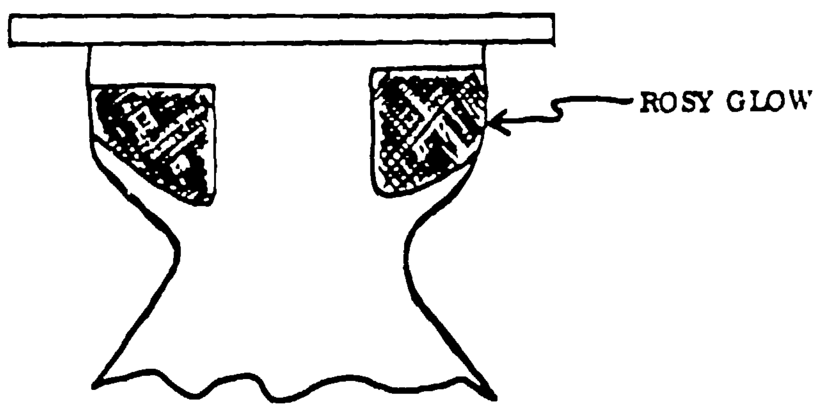
FIGURE 11

VIII. COMBUSTION STABILITY

Some concern was expressed in the Phase I study that the tuning of the acoustic cavities which are used to prevent combustion instability might change enough, due to the slightly different combustion temperature of hydrazine, to render them ineffective. Proper tuning occurs when the cavity resonant frequency matches the combustion chamber resonant frequency, in this case the first tangential acoustic mode of the chamber. For the MMH fuel thruster, this frequency is approximately 6000 Hz. Burn-through of the coated columbium chamber occurs rapidly (in three seconds) when unstable operation occurs.

To prevent damage to the borrowed thruster during the tests, a number of precautionary measures were taken. First, slower-acting facility valves were used instead of the regular fast-acting engine valves, since previous test experience indicated that the ignition explosion is an excellent trigger for combustion instability. Secondly, an accelerometer was mounted on the injector head at a previously calibrated position to sense the occurrence of combustion instability. A special thruster valve shutdown circuit was used which would terminate combustion if the vibration level remained at 50 g's for 0.3 second. In the initial facility and thruster checkout tests, the accelerometer was found to be malfunctioning. This led to the installation of a high-response Kistler transducer to provide additional monitoring of the combustion oscillations.

Visual evidence of possible resonant burning occurred during the first day of testing. The thruster thermal behavior was initially explored by increasing the run time from two seconds to five seconds. At the end of the five-second thermal run, the thruster appeared to have "rosy cheeks". The sketch below illustrates the appearance of the thruster. A rosy glow was observed on each side of the combustion chamber near the head, indicating localized heating of chamber.



Another five-second run was made with the same result. Since the accelerometer was not functioning properly, no quantitative measurement of the magnitude or frequency of any oscillation could be obtained. No further testing was conducted until the instrumentation could be improved, since this heating pattern is typical of the first tangential mode of combustion instability.

In the second series of tests, which documented the performance and thermal behavior of the thruster, no evidence of "rosy cheeks" was observed for either the same operating conditions or at any other operating condition in the test plan. Detailed examination of the high response accelerometer output and Kistler pressure revealed that all high frequency pressure disturbances were rapidly damped. This indicates that the acoustic cavities were providing effective damping. However, in the last run of the second test series (Run 36), the accelerometer signaled the presence of large amplitude oscillations at 5300 Hz at the very end of Run 36, as shown in Figure 12. Examination of the lower frequency response oscillograph data (Figure 13) indicates that severe perturbations of chamber pressure triggered the high frequency. This behavior of the chamber pressure is characteristic of what happens to combustion when large gas bubbles are passing through the propellant lines. The oxidizer manifold pressure and flowmeter traces indicated depletion of oxidizer. The absence of water hammer waves in the oxidizer lines (Figure 13) after closing of the oxidizer valve, provides similar evidence. Calculations of the amount of propellant consumed in the tests also indicated that the oxidizer tank began emptying near the end of the run. This caused the ingestion of the tank pressurant gases.

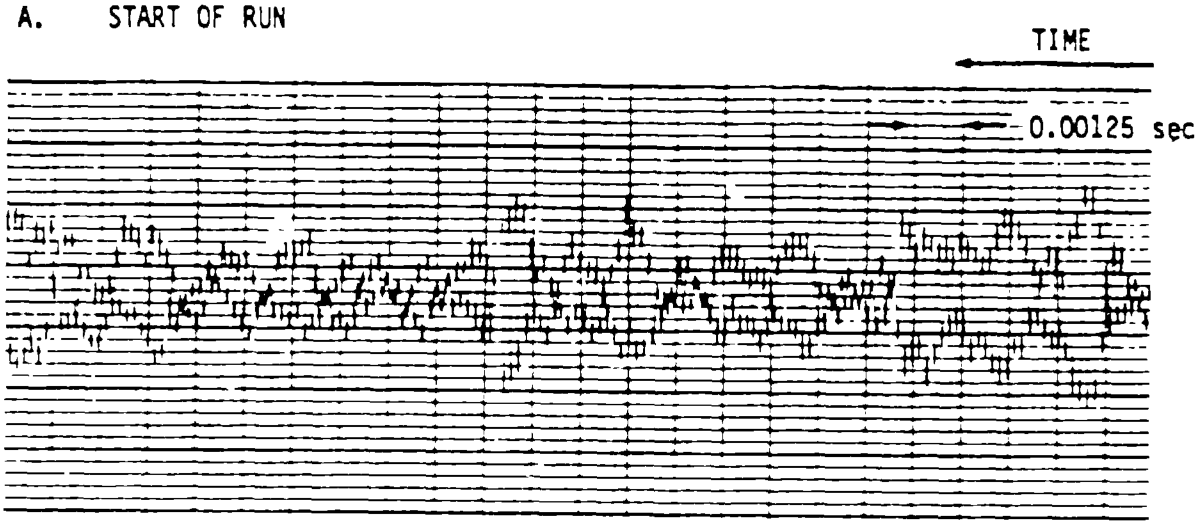
The bubbles in the oxidizer line produce many effects; first, the bouncing and oscillatory combustion shown in Figure 13. The induced combustion roughness acts to excite the resonant frequencies in the chamber. The depletion of oxidizer flow reduced the effective mixture from the mixture ratio of 1.0 targeted for the run to a mixture ratio estimated to be less than 0.5. The far off-design mixture ratio acts to change the damping of the acoustic cavities by changing the cavity gas temperatures. The behavior of the thruster during this portion of the run is not representative of either nominal or normal off-design operational conditions that the thruster would be subjected to in flight. Because oxidizer depletion and pressurant gas flow is not representative of normal operational conditions, the observed resonant burning is not considered a valid cause for concern. Experience with acoustic cavities has demonstrated that there is always some far off-design condition where the acoustic cavity resonant frequency is not matched to the chamber resonant frequency, and the acoustic damping is reduced.

The detection of resonant burning under these unusual test conditions emphasizes the need to document the combustion stability of the hydrazine-fueled Shuttle Thruster over the full range of normal off-design operating conditions. This was not done in this portion of the test program.

THRUSTER 2-4 - RUN 36

PLAYBACK OF ACCELEROMETER MAGNETIC RCORDING

A. START OF RUN



B. END OF RUN - OXIDIZ"R DEPLETION AND GAS BUBBLES

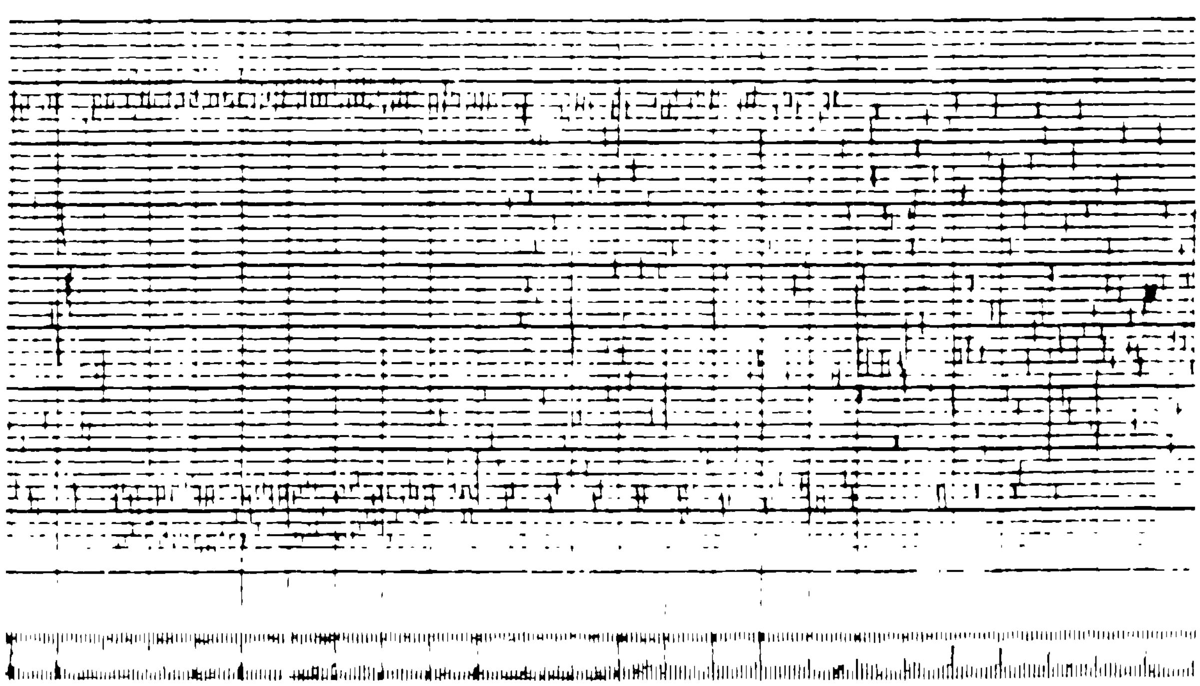


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FIGURE 12

THRUSTER 2-4
BEHAVIOR AT OXIDIZER DEPLETION

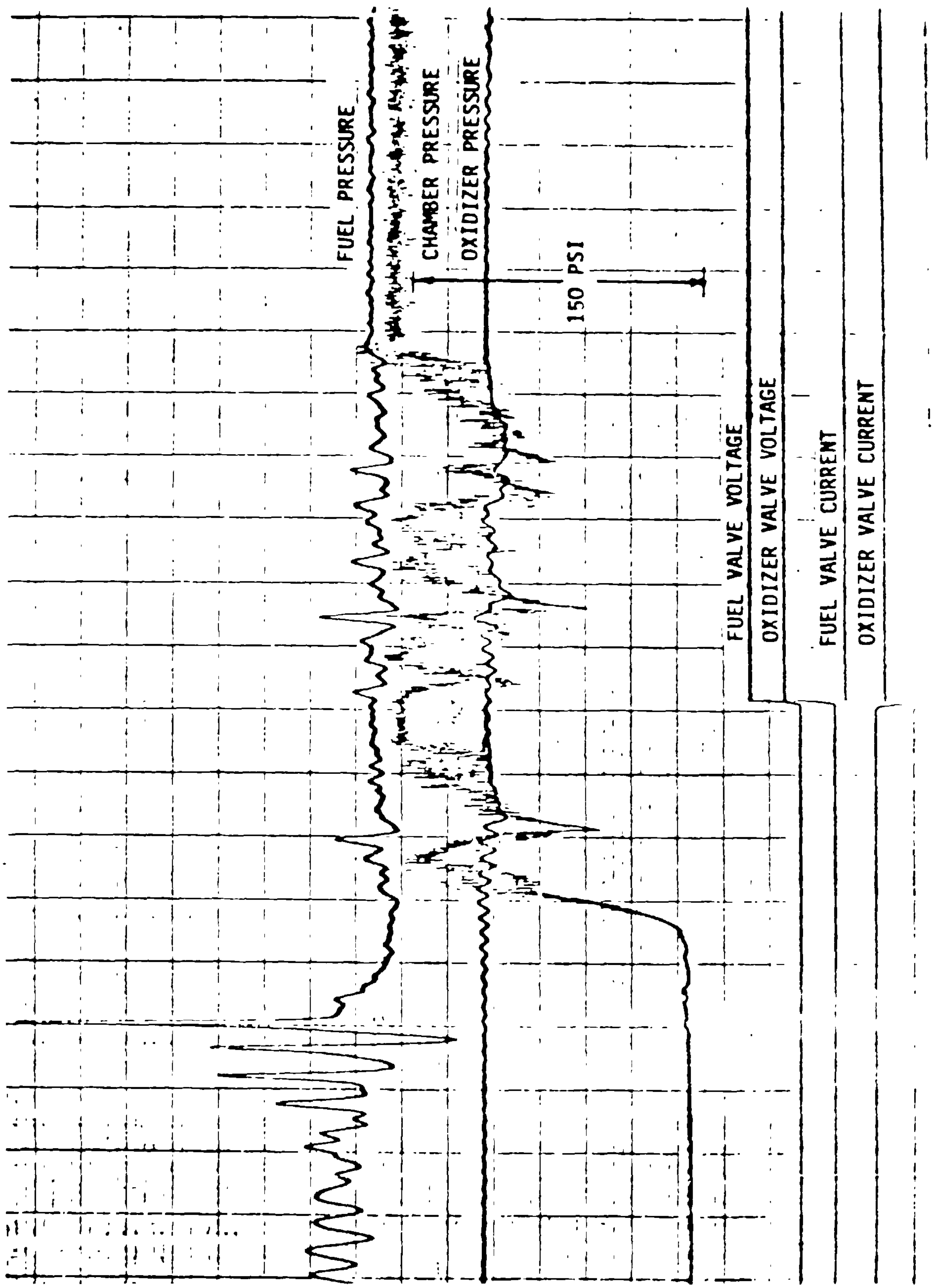

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FIGURE 13

IX. CONCLUSIONS

The tests of the Space Shuttle Reaction Control Subsystem thruster indicate that its performance and thermal characteristics with hydrazine as a fuel are equal to or better than that measured with MMH. Characteristic velocity at a mixture ratio of 1.4 (equal volumetric flow) for hydrazine was 5180 feet/second compared with 5110 feet/second at a mixture of 1.6 for MMH. Specific impulse with a 22:1 nozzle is calculated to be 280 lbf-sec/lbm.

The unexpected result of the testing was a throat temperature of less than 800°F for hydrazine at an O/F = 1.4 compared with a throat temperature of 1500°F for MMH at an O/F = 1.6. These temperatures were measured on a radiation-cooled chamber in both cases (i.e., with the normal SSRCS package removed). The predicted temperature for hydrazine at O/F = 1.4 was 2330°F, assuming a reactive film coolant.

The test data for the shuttle thruster with hydrazine as a fuel suggest that a transition between inert liquid cooling and reactive liquid film cooling occurs at mixture ratios greater than 1.6. The parameters governing the transition were not defined (other than the obvious effect of mixture ratio) by either the test program or the initial analytical work leading to the reactive film model.

Further test and analytical work would be desirable to evaluate the thruster high O/F thermal characteristics and define the transition O/F and mechanism.

The thruster exhibited no combustion instability over the range of normal test operating conditions. However, stability was not experimentally evaluated over the full range of potential flight off-design conditions.

X. RECOMMENDATIONS

Additional analytical and experimental work is required to define the behavior of liquid film cooling in rockets utilizing hydrazine as the coolant. The present test firing program provided data which seems to indicate that hydrazine as a film coolant can sometimes act as an inert coolant and at other times like a reactive coolant. The factors governing the different behavior modes must still be defined.

XI. REFERENCES

1. Minton, S.J., "Evaluation of the SSRCS Engine With Hydrazine as a Fuel", Phase I, The Marquardt Company Report S1423, January 1978.

2. Pfeifer, G.R., "Data Summary Report, Thruster 2-4 Performance Test", The Marquardt Company Report TM05-A020, 30 June 1976.

3. Roberson, J.A., "Engine 2-4 Performance Test", NASA/Lyndon B. Johnson Space Center, Doc. No. JSC-12895, 27 June 1977.

4. Antoine, A.C., "Effect of Pressure on Rate of Burning (Decomposition with Flame) of Liquid Hydrazine", NASA TN D-2694.

5. Rosser, Jr., W.A., Peskin, R.L., "A Study, Decomposition Burning", Combustion and Flame, Vol. 10, June 1966.

6. Glassman, I., "Combustion Processes in Liquid Propellant Rocket Motors", Princeton University, Final Report, 1 September 63-31 August 64. (AF 49 (638)-1268).

TEST PLAN

THE Marquardt COMPANY

TEST PLAN

MTP	REV.
0254	A
ISSUED 7-20-77	REVISED 1-31-78

FORM TMC 1032 REV 8-71

TITLE PERFORMANCE EVALUATION OF HYDRAZINE FUEL WITH AN 870 LBF THRUSTER

I. TEST OBJECTIVE

The test program will document the specific impulse and thermal behavior of a Space Shuttle Reaction Control Subsystem -- Primary Thruster operating with nitrogen tetroxide (NTO) and hydrazine as propellants. This plan describes the tests to be performed on P/N X30850, S/N 001 (designated as Thruster No. 2-4), a prototype 870 lbf thruster. As Thruster No. 2-4 has been tested using NTO and monomethylhydrazine (MMH) propellants, direct comparisons of documented parameters obtained with fuel as the only variable can be made.

Specific objectives of the test program include:

- Documentation of thruster performance characteristics at nominal operating conditions.
- Documentation of thruster performance characteristics over the operating range to obtain I_{sp}, C*, and O/F.
- Documentation of thruster thermal response during and after steady state operation at varying inlet pressures.

II. SCOPE

This plan delineates testing to be performed on Thruster No. 2-4. The plan provides for testing the thruster with hydrazine fuel rather than monomethylhydrazine. Testing will be conducted per the following sequence in Marquardt's Pad F.

Bolted ϵ = 22:1 Chamber - Uninsulated

Appendix	Test	Page
A	Baseline Performance	A-1
B	Steady State Performance	B-1
C	Thermal Characterization	C-1

III. GENERAL CONSIDERATIONS

A. Thruster Description

1 Injector Design:

P/N 30850
84 doublet dual momentum angle.
Single ring with 24.9% film cooling.

Nominal injector hole sizes:

	Inner	Outer
Oxidizer, Main	0.0315	0.0315
Fuel, Main	0.0225	0.0225
Fuel Cooling 20°	0.020	--
Fuel Cooling 45°	--	0.0165

2. Nominal Thrust 870 lbf.

3. Mixture Ratio 1.600 ± .032 (for MMH)

4. Valves - Line Fluid Actuated

5. Bolted Thrust Chamber, P/N X30958, and Exit Bell, P/N X30084, Expansion Ratio = 22:1, Throat Dia. = 2.050 in., Exit Dia. = 9.6 in.

B. Instrumentation

The basic instrumentation for hot firing tests is shown in Table I. Test data will be acquired by the Pad F Instrumentation. Data will be recorded on Bristols and the oscillograph. Each hot firing test plan appendix will specify any additional instrumentation and/or deletions required for the specific tests.

C. Test Deviations/Discrepancies

In the event of failure of the test item, testing shall be stopped and the Project Engineering, Manager of Development Engineering, and the Reliability Engineer shall be notified. Failure investigation shall be performed within the guidelines plan content changes from the approved test plan will be documented by a test deviation sheet (Figure 1) that will be initiated by the Development Engineer and approved by the Project Engineer and the Jet Propulsion Laboratory (JPL) Representative.

Minor procedural changes or reruns required because of normal facility problems; i.e., failure to obtain required pressure conditions, instrumentation failure, etc., may be directed by the Development Engineer at his discretion. Reasons for the changes/reruns will be recorded in the Development Engineering Thruster Test Logbook.

D. Test Fluids

Fuel - Hydrazine	MIL-P-26536, Amendment 1
Oxidizer - Nitrogen Tetroxide (NTO)	per SE-S-0073B (MON-3)
Pressurant - Helium (He)	per SE-S-0073B
Purging Agent - Nitrogen (N_2)	per SE-S-0073B
Flushing Agent - Reagent Water (H_2O)	per SE-S-0073B

E. Thruster Startup and Shutdown Procedure Pad F - Schematic, Figure 1

The following prime and purge procedure shall be used for these tests:

Prime

1. Open bypass valves to release trapped GN_2.
2. Reduce GN_2 purge pressures to 25 psig.
3. Close purge valves - manifolds will decay to check valve pressure.
4. Close bypass valves.
5. Reduce tank pressures to 80 psig, using slow bleeds.
6. Open main shutoff valves (Annin) to prime.
7. Bypass for 3 seconds.
8. Increase tank pressures to run set points.

Purge

1. Close main shutoffs (Annin) at test stand.
2. Set GN_2 purge at 200 PSI.
3. Open engine bypass valves (lines go to scrubbers and have check valves requiring 10 to 25 PSI to open).
4. Open manifold purge valves, (pushes propellants out of manifolds) for 30 seconds, then cycle 5 sec. off - 30 sec. on 5 times. At the conclusion of the cycling, the bypass valve will be closed, and the purge valve open, trapping the GN_2 at 200 PSI.
5. Select pulses to oxid and open engine valve.
6. Cycle oxid purge valve 15 sec. on, 15 sec. off 5 times. Purge pressure will be trapped in manifold.
7. Repeat 5 & 6 for fuel.

F. Inspection

Thruster firing shall be per facility operating procedure.

No inspection is required. The Development Engineer will be responsible for thruster development logbooks and the "Quick Look Run Data."

G. Documentation

A log shall be maintained of all testing on each engine assembly. These records shall be incorporated into the buildup logbook to form a complete engine history. Documents shall include, but not be limited to, the shop traveler, deviation sheets, engineering orders, etc.

H. Post Test Requirements

1. Purge the thruster per Paragraph III-E of MTP 0254 after each day of operation and before removal from test cell when required.

2. After removal from the test cell water-flush and GN_2 purge the oxidizer and fuel valves per the following:

	Pressure - psig	ON Time - Sec.	OFF Time - Sec.	No. of Cycles
Water	80	1	1	50
Nitrogen	300	1	1	50

3. The thruster is to be dried, after water-flush, by heating for a minimum of 1/2 hour at 170 ± 30° F at ambient pressure, then for a minimum of 3 hours at a temperature of 170 ± 30° F and ambient pressure of 0.3 or less.

I. Handling and Contamination Control

1. Piggyback filters with a rating of 25 microns absolute shall be used in all hot firing tests and during flush and purge operations.

2. During storage, transportation, and whenever possible during handling, the thruster is to be kept in a clean plastic bag.

3. A protective cover will be maintained on the engine valves at all times other than actual testing or oven drying.

4. Krytox shall be used on the assembly inlets prior to installation on the valves or the inlet lines per MPS 1103.

TABLE I

INSTRUMENTATION

Parameter	Symbol	Range	Units	OSC	Bristol	Other
Pressure, Oxidizer Manifold	P_{mo}	0–500	psig	X	X	
Pressure, Fuel Manifold	P_{mf}	0–500	psig	X	X	
Pressure, Chamber	P_c	0–200	psia	X	X	
Pressure, Cell Altitude	P_{cell}	0–40	psia	X	X	Gage
Flow Rate, Oxidizer	W_o	0–3.0 lb/sec	Hz	X	X	
Flow Rate, Fuel	W_f	0–2.2 lb/sec	Hz	X	X	
Temperature, Ox Flowmeter	T_{fmo}	0–200	^oF		X	
Temperature, Fuel Flowmeter	T_{fmf}	0–200	^oF		X	
Current, Oxidizer Valve	I_o	0–2	amp	X		
Current, Fuel Valve	I_f	0–2	amp	X		
Voltage, Oxidizer Valve	VO	0–32	volts dc	X		
Voltage, Fuel Valve	V_f	0–32	volts dc	X		
Temperature, Injector Head	T_h, T_{h3}	0–500	^oF		X	
Temperature – Injector Flange	$T_f, T_{f2}, T_{f3}, T_{f4}$	0–500	^oF		X	
*Temperature – Chamber	T_7	0–3000	^oF		X	
*Temperature–Chamber	T_8	0–3000	^oF		X	
*Temperature – Chamber	T_9	0–3000	^oF		X	
*Temperature – Chamber Throat	$T_{10}, T_{22}, T_{23}, T_{24}$	0–2500	^oF		X	
** Accelerometer – Cutoff	Aco	0–100	g			FM Tape

* Cutoff at 2600^{o}F at any location

** Cutoff set at 25 g.

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TEST PROGRAM DEVIATION/DISCREPANCY SHEET

DATE ______________________

NUMBER ______________________

APPLICABILITY:

MTP ______________ MIN ______________ TDP ______________

PROGRAM ____

THRUSTOR/ENGINE P/N ______________________ S/N ______________

FACILITY ______________________ ORIGINATOR ______________

APPROVALS (as required by Test Plan)

1. ______________________ 4. ______________________
2. ______________________ 5. ______________________
3. ______________________ 6. ______________________

TYPE:

☐ Discrepancy ☐ Procedure Change ☐ Document Change

Explain discrepancy or change and its effect: ______________________

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Explain reason for change or corrective action: ______________________

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DISTRIBUTION (Per Test Plan):

☐ Original to Change Control	☒ Project Engineer (2)	☐ DCAS Office
☒ Original to Development Test	☒ Data Analysis	☐ Customer
☒ Development Test Logbook	☐ Test Program Manager	☒ Reliability
☒ Thrustor/Engine Logbook	☐ Test Operations Engr.	☐ ______________

Figure 1

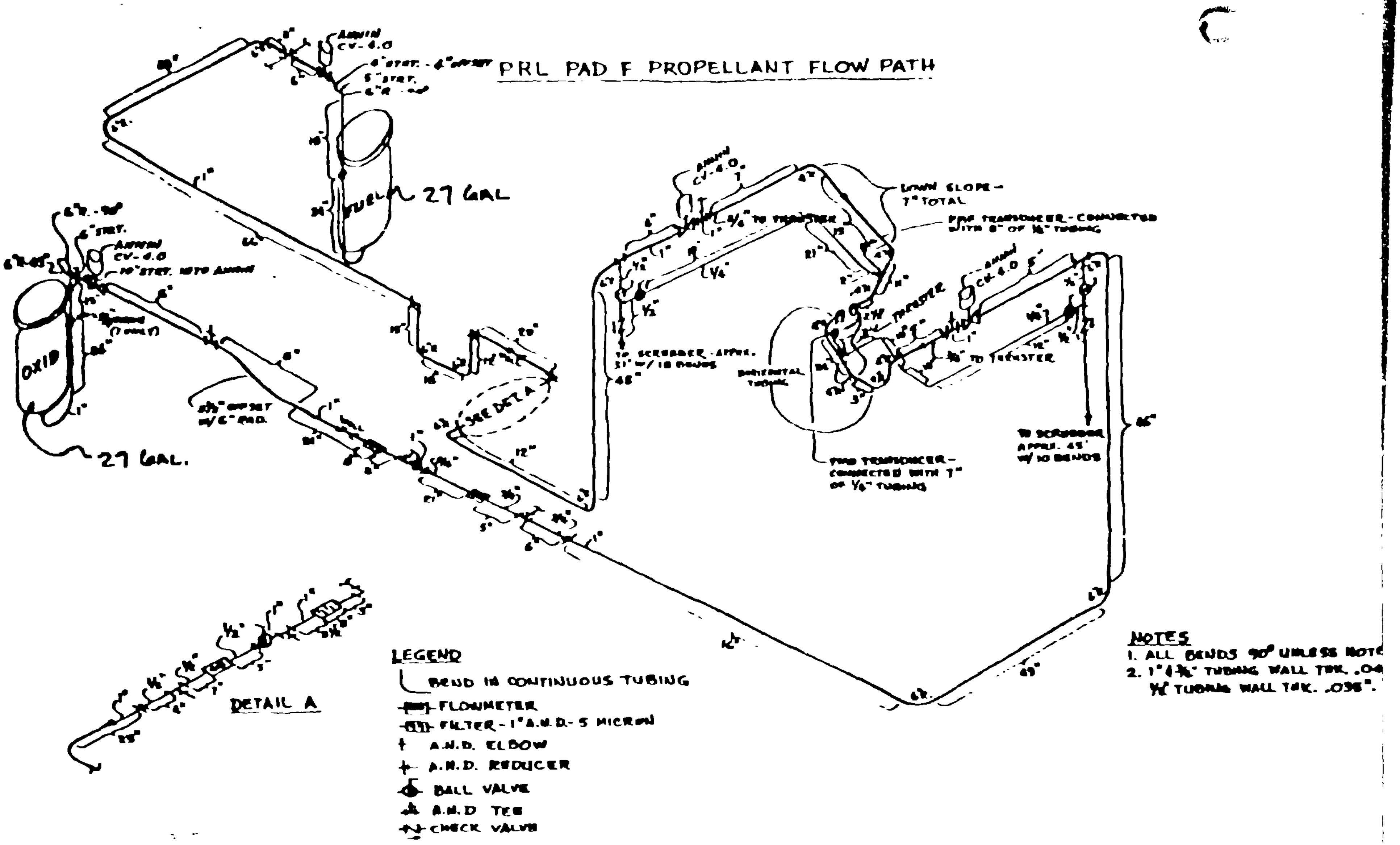

FIGURE 2

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APPENDIX A

BASE LINE PERFORMANCE

1.0 OBJECTIVE

This test provides a series of identical firings to define an acceptable level of thruster performance.

2.0 SCOPE

These tests will document the steady state performance at the nominal operating conditions using nonsaturated propellant.

3.0 DATA REDUCTION

The following data shall be obtained:

A. Steady State

1. Characteristic velocity
2. Mixture ratio
3. Propellant flow rates
4. Propellant temperatures
5. Chamber pressure
6. Dynamic inlet pressure
7. Specific impulse (computed from previously measured C_F)
8. Combustor and injector temperature -- for 30-second run only.

4.0 INSTRUMENTATION

The instrumentation shall be per Table I plus:

XR and vericolor photographs shall be taken during the 28 $\pm$ 1 volt DC 30-second run only.

5.0 STANDARD CONDITIONS

Propellant temperature - Ambient

Valve voltage — 28 vdc

6.0 RUN SEQUENCE

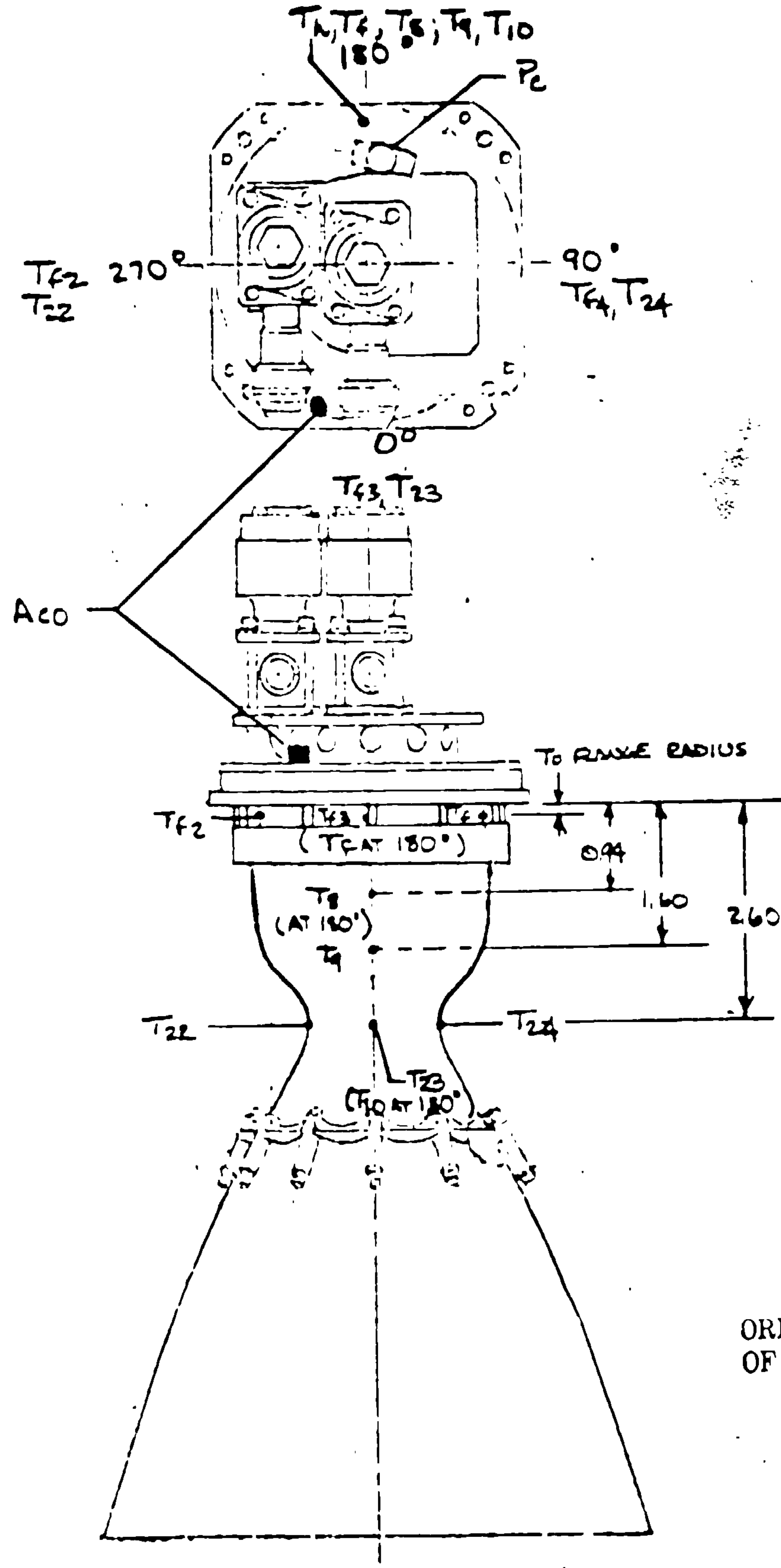
A minimum of two runs shall be made at these conditions:

Order	Target O/F (±.032)	Target Chamber Pressure	No. of Pulses	Duration Sec.	Pmo psig	Pmf psig
1	1.4	150	1	2.0	TBD	TBD
2	1.4	150	1	2.0		
3	1.4	150	1	30.0		

7.0 COMBUSTION INSTABILITY CONSIDERATIONS

An accelerometer is used to monitor the onset of combustion instability in the thruster. The output of the accelerometer is recorded on magnetic tape for subsequent analysis of its frequency spectrum. The output of the accelerometer is also fed to an automatic valve shutdown device which terminates thruster operation when the vibration level exceeds 25 g's. Vibration levels in excess of 5-15 g (normal operation) indicate the presence of combustion oscillations (generally the first tangential combustion instability) which cause excessive heating of the chamber.

In the event that combustion instability is detected, under some operating conditions, long duration runs (in excess of 4-5 secs) will be eliminated to avoid chamber burnthrough.



INSTRUMENTATION LOCATIONS

APPENDIX B

STEADY STATE PERFORMANCE

1.0 OBJECTIVE

This test will provide thruster performance characteristics over a range of test variables. The data obtained from these tests will be used in a program to correlate thruster performance parameters.

2.0 SCOPE

A series of steady state and pulse firings will be conducted on the thruster at various inlet pressures with uncontrolled propellant saturation to provide the necessary data.

3.0 DATA REDUCTION

The following data shall be obtained:

A. Steady State

1. Characteristic velocity
2. Mixture ratio
3. Propellant flow rates
4. Propellant temperatures
5. Chamber pressure
6. Dynamic inlet pressure
7. Specific impulse (computed from previously measured C_F)
8. Combustor and injector temperature

4.0 INSTRUMENTATION

The instrumentation shall be per Table I.

5.0 STANDARD CONDITIONS

1.	Propellant temperature	Ambient
2.	Propellant saturation	Uncontrolled
3.	Inlet Pressure	As required
4.	Voltage	28 vdc

6.0 RUN SEQUENCE

Steady State Firing

Two firings per condition - five-second firings (trim runs may be made as required to obtain run conditions).

No.	Estimated Inlet Set Pressure (psig) PMO	PMF	Target O/F ±0.05	Target Chamber Pressure ± 10 PSIA
1	251	243	1.4	150
2	245	250	1.3	150
3	262	238	1.5	150
4	289	227	1.8	150
5	372	221	2.4	150
6	224	281	1.0	150
7	212	322	0.8	150
8	251	243	1.4	150

APPENDIX C

THERMAL CHARACTERIZATION
(Welded 22:1 Chamber)

1.0 OBJECTIVE

The purpose of these tests is to document the temperature response and margins of the thruster during steady state operation. Tests will be conducted over a range of propellant pressures to thermal equilibrium and maximum soakback.

2.0 SCOPE

The thruster will be instrumented and operated for extended run lengths to define the impact of long run times upon thruster performance and temperature structural analysis.

3.0 DATA REDUCTION

XR and vericolor photographs shall be taken during the 50-second runs at 10-second intervals. Time plots will be prepared for:

1- Thruster Thermocouples (run and soakback)

3 - P_c Run only

4 - I_{sp} Run only

Date will be taken until the end of the run, and until the end of soakback on 50 second runs.

4.0 INSTRUMENTATION

See Figure C-4 for thermocouple locations. Instrumentation shall be per Table C-1.

5.0 STANDARD CONDITIONS

All runs 50 seconds. Soakback times of 5-10 minutes or until peak injector temperatures are established will be recorded on at least one run to allow maximum injector temperatures to be achieved.

Any run is to be terminated if any temperature exceeds 2600°F.

Propellant saturation is uncontrolled.

Propellant temperature ambient.

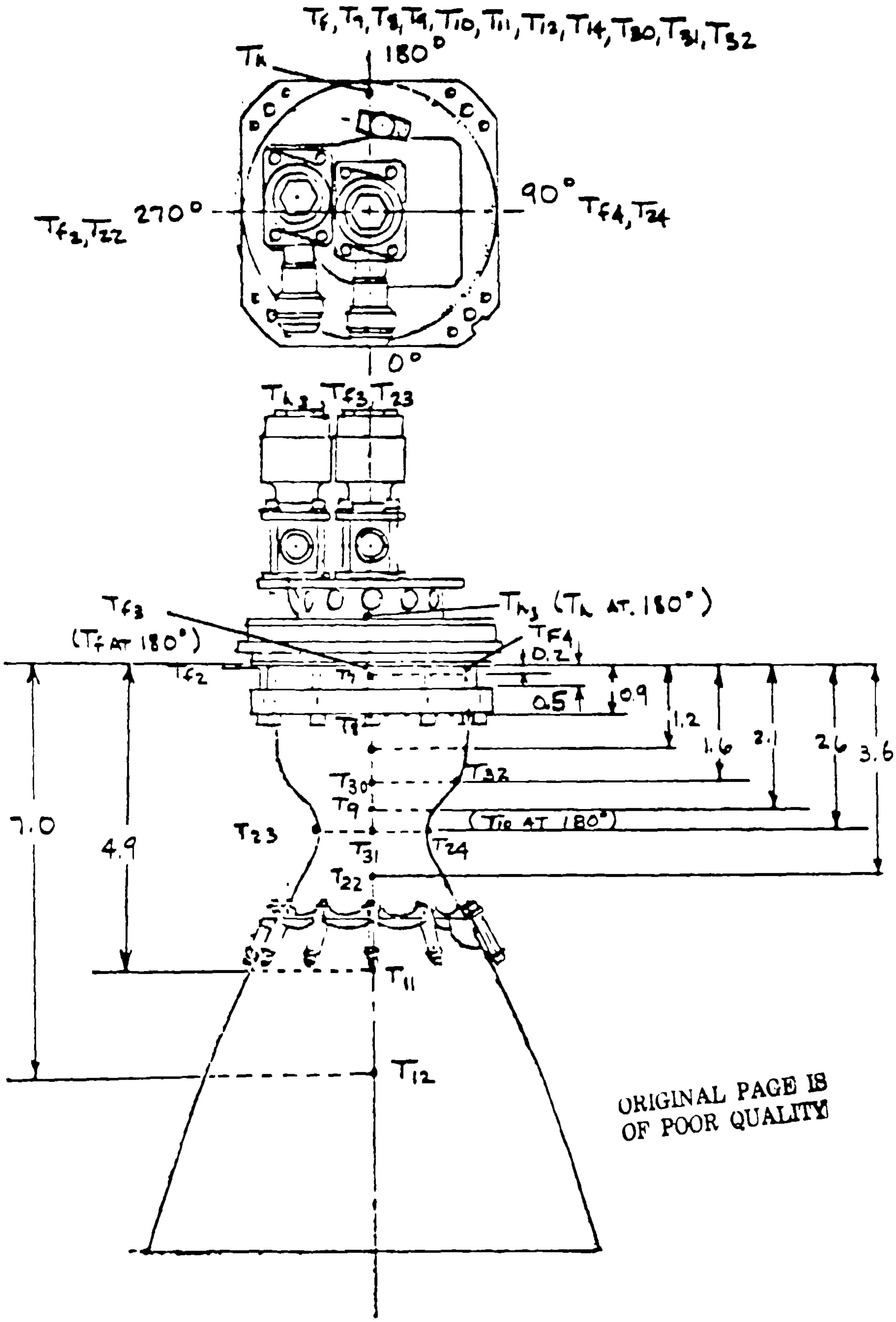
TABLE C - I

CELL INSTRUMENTATION

	Symbol	Range	Units	OSC	Bristol	Other
Pressure, Manifold, Ox.	P_{mo}	0-500	psig	X	X	
Pressure, Manifold, Fuel	P_{mf}	0-500	psig	X	X	
Pressure, Chamber	P_c	0-200	psia	X	X	
Pressure, Cell	P_{cell}	0-0.4	psia	X		
Flow Rate, Ox.	W_o	0-3.0 pps	Hz(vdc)	X	X	
Flow Rate, Fuel	W_f	0-2.2 pps	Hz(vdc)	X	X	
Temperature, Ox. Flowmeter	T_{fmo}	0-200	°F		X	
Temperature, Fuel Flowmeter	T_{fmf}	0-200	°F		X	
Temperature, Injector Flange	T_f	0-500	°F		X	
Temperature, Injector Flange	T_{f_2}	0-500	°F		X	
Temperature, Injector Flange	T_{f_3}	0-500	°F		X	
Temperature, Injector Flange	T_{f_4}	0-500	°F		X	
Temperature, Injector Head	T_h	0-500	°F		X	
Temperature, Injector Head	T_{h_3}	0-500	°F		X	
*Temperature, Chamber Wall	T_7	0-2500	°F		X	
*Temperature, Chamber Wall	T_{30}	0-2500	°F		X	
*Temperature, Chamber Wall	T_8	0-2500	°F		X	
*Temperature, Chamber Wall	T_{31}	0-2500	°F		X	
*Temperature, Chamber Wall	T_9	0-2500	°F		X	
*Temperature, Chamber Throat	T_{10}	0-3000	°F		X	
*Temperature, Chamber Throat	T_{22}	0-3000	°F		X	
*Temperature, Chamber Throat	T_{23}	0-3000	°F		X	
*Temperature, Chamber Throat	T_{24}	0-3000	°F		X	
*Temperature, Nozzle	T_{11}	0-3000	°F		X	
*Temperature, Nozzle	T_{12}	0-3000	°F		X	
Current, Thruster Ox. Falve	IO	0-2	amp	X		
Current, Thruster Fuel Valve	IF	0-2	amp	X		
Voltage, Thruster Ox. Valve	VO	0-32	VDC			
**Accelerometer, Cutoff	ACO	0-100	g's			FM Tape, Pulser

* Cutoff at 2600°F any location.

** Accelerometer stop run (pulser cutoff) when "G" level exceeds 25 g's.



THERMOCOUPLE LOCATIONS

Figure C-4

6.0 RUN SEQUENCE

Run one firing at each point except as needed. Run duration to be 50 seconds. Trim runs may be made (2.0 - second duration) to establish run conditions.

Order	Estimated Inlet Set Pressures (psig)		O/F ± 0.05	Estimated Chamber Pressure (psia)
	PMO	PMF		
1	251	243	1.4	150
2	212	322	0.8	150
3	372	221	2.4	150

TABULATED DATA

QUICK LOOK RUN DATA

X30850	001	Thruster No. 2-4	Pod F	MTP 0254	3812	1-31-78	L. LARSON	S. Murray, D. Rupple, S. Smith
N_2H_4	FAC 228 01-12-78		NTO	FCC 353 01-12-78		C-103	2.0905" I.D. 3.3012 in^2	4.6" I.D. 1795

Run									Remarks
1	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	Data reduced from [illegible] and calc. made
2	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
3	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	Using Cell 1795 (constant)
4	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
5	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
6	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
7	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
8	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
9	[illegible]	5.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	[illegible] loop [illegible] at 14.6 sec.
10	[illegible]	5.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
11	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	
12	[illegible]	2.0	[illegible]	[illegible]	14.	[illegible]	[illegible]	[illegible]	

Run 10

QUICK LOOK RUN DATA

~ 24 psia
Cell Pressure
with insulation package

X30850	001	2-4	Pad F	MTP 0254	3812	2-22-78	L. Lapsin	S. Morton, S. Smith
N_2H_4	FCC 228 / 1-16-78	NTO	FCC 353 / 1-12-78	C-105	3.292	9.6 in. I.D.	1.735	

Run	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	Notes	[illegible]	[illegible]	Remarks
13	[illegible]	1.0	[illegible]	[illegible]	15.1	6.5		440	1412	Orifice [illegible] in. ID; 0.0 [illegible]; 10; 0.0001400
14	[illegible]	2.0	[illegible]	[illegible]	[illegible]	6.5		440	1500	
15	[illegible]	2.0	[illegible]	[illegible]	[illegible]	6.5		578	1500	
16	[illegible]	2.0	[illegible]	[illegible]	[illegible]			464	[illegible]	
17	[illegible]	2.0	[illegible]	[illegible]	15.0			487	1300	
18	[illegible]	2.0	[illegible]	[illegible]	[illegible]			465	1212	
19	[illegible]	2.0	[illegible]	[illegible]	15.0	7.0		472	1198	
20	[illegible]	2.0	[illegible]	[illegible]	[illegible]	7.0		570	1432	
21	[illegible]	1.0	[illegible]	[illegible]	[illegible]			395	1430	
22	[illegible]	1.0	[illegible]	[illegible]	[illegible]	6	[illegible]	310	1380	
23	[illegible]	1.0	[illegible]	[illegible]	[illegible]	6		440	1440	[illegible]
24	[illegible]	2.0	[illegible]	[illegible]	[illegible]			538	1135	
25	[illegible]	2.0	[illegible]	[illegible]	[illegible]			440	1338	
26	[illegible]	2.0	[illegible]	[illegible]	[illegible]			530	1213	
27	[illegible]	[illegible]	[illegible]	[illegible]			Roll & Cutoff			[illegible]
28	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		Pulse at 33.5 sec.	380	1430	
29	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		[illegible] 10 sec.	385	1442	[illegible]
30	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		Pulse at 4 sec.	495	1380	
31	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		Pulse at 5.8 [illegible]	465	1400	
32	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]		Pulse at [illegible] 30	460	1405	

QUICK LOOK RUN DATA

P-2

33	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	530	700		
34	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	525	700		
35	[illegible]	[illegible]	270	[illegible]	[illegible]	[illegible]	655	[illegible]		[illegible]
36	[illegible]	[illegible]	270	310	[illegible]	[illegible]	580	[illegible]	[illegible]	

GENERAL CALCULATION SHEET

MAC [illegible] REV. 4-57

AMBIENT SITE CONDITIONS FUEL PERFORMANCE DATA AT= 3.292

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
RUN No.	PMO	PMF	Pc	DPOX	DPF	WO	WF	O/F	WT	TFMO	TFMF	CSTAR	ISP	FT	CF
#	PSIA	PSIA	PSIA	PSID	PSID	LB/SEC	LB/SEC		LB/SEC	°F	°F	FT/SEC	SEC	LBF	
1.	215.9	244.0	148.4	67.9	96.0	2.180	1.142	1.909	3.322	54.8	55.2	4735	255.1	847	
2.	194.1	244.0	144.6	54.4	104.3	1.884	1.162	1.621	3.046	63.3	57.6	5032	271.1	826	
3.	194.1	252.5	146.0	53.9	112.3	1.853	1.170	1.584	3.022	62.8	55.6	5121	275.9	834	
4.	194.6	266.0	148.8	51.8	83.2	1.811	1.250	1.449	3.061	60.1	56.0	5152	277.6	850	
5.	194.6	282.5	151.8	48.2	136.1	1.786	1.320	1.353	3.106	56.4	56.4	5180	279.1	867	
6.	189.4	296.5	151.6	40.2	147.3	1.671	1.392	1.201	3.064	53.5	54.2	5245	282.6	866	
7.	182.7	314.3	151.2	38.9	170.5	1.562	1.483	1.053	3.045	59.1	55.4	5264	283.6	864	
8.	175.2	337.9	152.2	32.0	194.7	1.443	1.570	0.919	3.013	59.7	53.5	5355	288.5	869	
9.	197.5	281.0	152.0	50.1	133.6	1.803	1.322	1.364	3.125	55.4	56.0	5155	277.8	868	
10.	200.0	282.0	153.6	56.4	138.4	1.812	1.314	1.379	3.126	59.3	55.7	5206	280.6	877	
11.	216.4	243.3	148.2	67.8	94.7	2.174	1.138	1.911	3.312	54.2	53.3	4743	255.5	846	
12.	237.7	244.0	151.4	89.4	95.7	2.432	1.121	2.170	3.553	54.5	54.4	4517	243.4	865	

GENE. ALCULATION SHEET

MAC 0030 REV. 4-67

AMBIENT SITE CONDITIONS — FINAL PERFORMANCE DATA

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
Run No.	P_{MO}	P_{MF}	P_C	D_{POX}	D_{PF}	W_O	W_F	O/F	W_T	T_{FMO}	T_{FMF}	CSTAR	ISP	F_T
#	PSIA	PSIA	PSIA	PSID	PSID	LB/SEC	LB/SEC		LB'S-C	°F	°F	FT/SEC	SEC	LBF
13.	234.0	282.9	150.0	84.0	132.9	1.728	1.323	1.306	3.051	73.2	66.9	5207	280.8	857
14.	231.0	288.2	150.8	80.2	137.4	1.706	1.345	1.268	3.051	71.5	66.9	5235	282.3	861
15.	202.0	290.7	143.7	58.3	147.0	1.466	1.402	1.046	2.868	71.5	66.8	5307	286.2	821
16.	255.8	258.4	146.6	89.2	111.8	1.801	1.226	1.469	3.027	77.9	64.7	5130	276.6	837
17.	250.0	262.5	149.7	100.3	112.8	1.919	1.233	1.556	3.152	70.3	64.3	5030	271.2	855
18.	305.1	236.8	147.9	157.2	88.9	2.368	1.095	2.163	3.483	67.4	62.2	4523	243.9	845
19.	221.0	290.9	149.8	71.2	141.1	1.635	1.363	1.200	2.998	67.6	63.4	5292	285.4	856
20.	208.5	312.4	148.4	60.1	164.0	1.484	1.466	1.012	2.950	69.7	64.9	5328	287.3	848
21.	202.5	323.7	147.8	54.7	175.9	1.436	1.507	0.953	2.943	70.8	65.5	5319	286.8	844
22.	187.0	347.0	146.2	40.8	200.8	1.257	1.615	0.778	2.872	91.3	61.2	5392	290.8	835
23.	236.0	276.1	150.9	85.2	125.3	1.773	1.286	1.379	3.059	73.8	68.2	5221	281.5	861
24.	320.5	229.7	146.1	174.4	83.6	2.517	1.063	2.368	3.580	77.1	63.5	4322	233.1	834
25.	252.5	256.2	149.6	102.9	106.6	1.957	1.198	1.634	3.155	79.4	63.5	5022	270.8	854
26.	260.0	241.8	146.2	113.8	95.6	2.046	1.138	1.798	3.184	78.8	65.7	4863	262.2	835
~~27.~~	~~208.0~~	~~313.1~~	~~148.8~~	~~65.2~~	~~190.3~~	~~1.518~~	~~1.461~~	~~1.046~~	~~2.969~~	~~73.9~~	~~68.2~~	~~5091~~	~~279.9~~	~~846~~
28.	207.5	308.1	148.4	59.1	159.7	1.492	1.449	1.030	2.941	70.3	59.2	5344	288.2	848
29.	211.5	319.7	150.9	60.7	168.9	1.508	1.480	1.019	2.988	63.5	57.1	5345	288.2	861
30.	253.5	229.8	152.6	99.9	115.2	1.948	1.244	1.566	3.192	64.5	57.3	5064	273.1	872
31.	240.5	270.3	151.0	89.5	119.3	1.830	1.259	1.454	3.089	62.5	58.0	5177	279.2	862
32.	236.5	270.5	150.4	86.0	120.1	1.806	1.264	1.429	3.070	62.5	58.1	5189	279.8	859
33.	265.0	247.1	147.4	117.6	99.7	2.075	1.151	1.803	3.226	62.5	57.6	4839	260.9	842
34.	263.5	246.9	147.4	116.1	99.5	2.079	1.152	1.805	3.231	62.9	58.8	4832	260.6	842
35.	327.8	225.7	144.2	183.6	81.5	2.591	1.051	2.465	3.642	63.3	58.7	4194	226.2	824
36.	209.3	319.7	150.4	56.7	167.1	1.517	1.466	1.035	2.983	64.1	56.6	5325	287.1	857

THERMAL DATA

Run No.	Total Seconds	Time Intervals	T_H	T_{H3}	T_F	T_{F3}	T_{F4}	T_7	T_8	T_9	T_{10}	T_{11}	T_{12}	T_{22}	T_{23}	T_{29}	T_{30}	T_{31}	T_{32}
28	5.07	Start	96	93	97	100	98	96	118	80	80	14		88	92	90	64	84	85
		5	145	93	126	133	147	125	175	308	273	208		306	290	105	68	266	85
29	10.2	Start	95	93	97	100	99	96	120	79	80	14		89	90	93	73	83	84
		5	132	93	121	131	145	120	155	313	268	212		395	279	95	73	270	84
		10	167	94	165	165	193	141	160	319	290	246		384	284	103	75	273	84
30	5.08	Start	113	102	114	115	119	105	123	85	88	15		88	95	93	83	95	83
		5	153	102	92	149	168	135	178	343	923	232		1403	945	98	178	700	83
31	10.0	Start	136	115	137	140	136	125	123	98	96	15		96	105	118	93	100	128
		5	169	115	163	204	187	145	200	328	625	239		1388	581	120	180	418	303
		10	190	116	216	203	243	168	205	335	955	268		1750	775	168	228	560	303
32	29.48	Start	107	101	109	108	107	123	120	83	83	14		88	95	105	78	88	368
		5	159	101	135	142	158	130	173	328	112	225		1450	488	143	165	405	468
		10	191	101	190	170	223	159	150	338	500	259		1755	704	173	223	518	495
		15	205	101	239	189	266	168	150	343	753	259		1795	813	198	258	595	511
		20	216	102	263	204	292	178	150	345	578	291		1808	913	223	280	658	515
		25	226	103	285	212	307	180	150	345	573	271		1805	989	245	288	695	518
		30	236	103	297	219	319	179	150	348	705	255		1803	1033	268	295	723	517
33	10	Start	129	112	129	129	127	114	120	88	88	16		91	100	148	80	95	105
		5	161	112	183	158	163	140	173	500	1050	230		1790	1440	170	188	1359	200
		10	179	112	200	187	219	165	150	550	1445	259		1805	1765	264	243	1491	205
34	29.14	Start	145	119	142	145	144	121	123	75	93	17		96	103	140	85	95	70
		5	172	119	165	179	191	143	163	465	955	231		1575	1348	145	175	1275	93
		10	191	120	206	200	236	166	168	523	1365	256		1880	1680	163	233	1558	95
		15	209	120	246	218	266	178	175	560	1508	246		1933	1795	185	263	1575	95
		20	223	120	278	235	286	180	175	570	1520	251		1946	1858	-109	280	1600	95
		29.14	234	120	300	240	[illegible]	188	175	578	1533	254		1963	1830	-65	295	1650	95
35	5.0	Start	136	117	135	134	134	118	128	88	88	16		93	99	105	80	95	93
		5	158	116	158	159	178	138	158	1720	1395	220		1523	1575	120	245	1644	93
36	19.1	Start	-24	100	110	109	106	96	118	75	78	15		79	88	133	62.5	79	93
		5	81	100	156	183	152	238	175	310	243	196		163	298	150	158	237	93
		10	320	100	199	210	196	225	188	315	248	229		291	318	148	193	246	93
		15	380	101	236	214	219	223	170	318	250	236		453	325	28	203	255	93
		19.1	418	101	261	256	245	293	190 (?)	320	255	233		768	319	-105	2?5	255	93
			BAD						BAD							BAD NEGATIVE TEMPS?			