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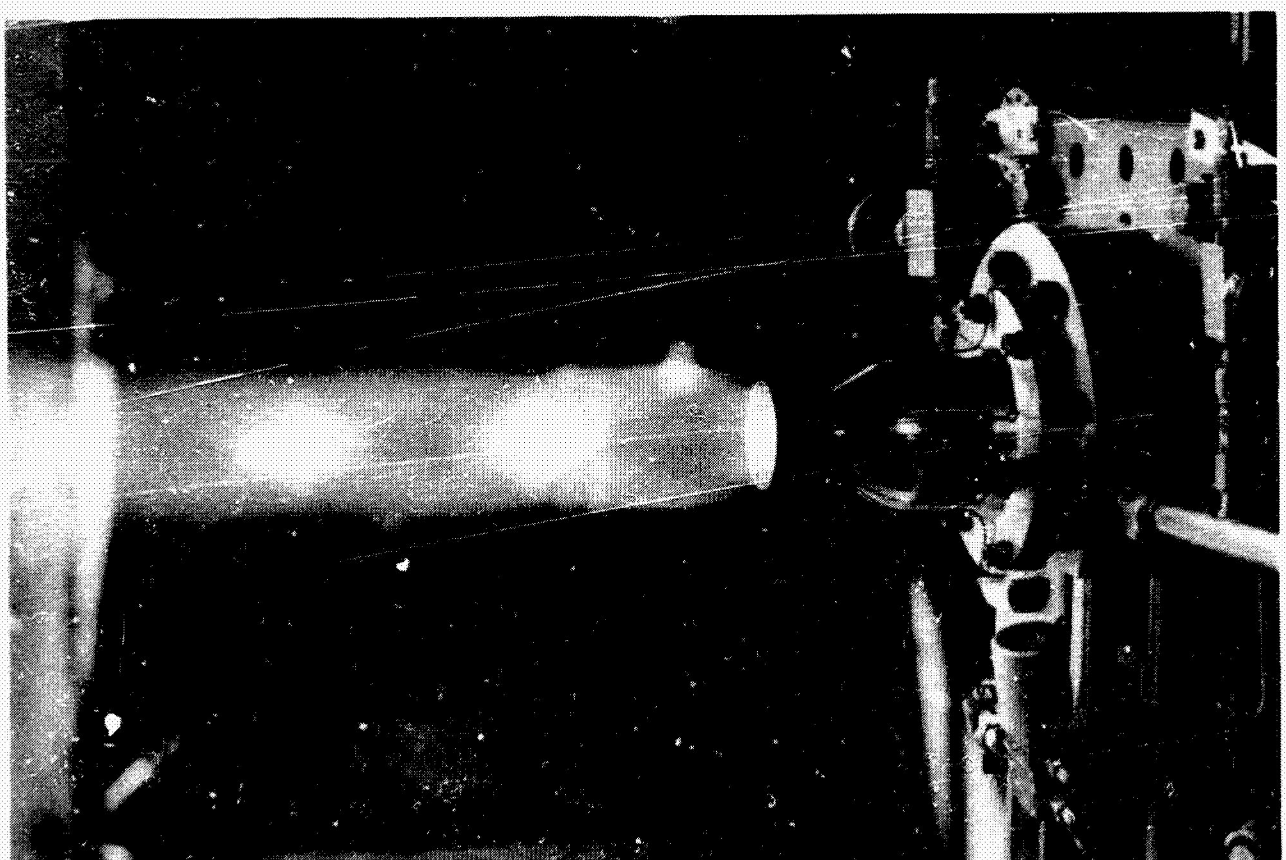
Ignition Characterization Of LOX/Hydrocarbon Propellants

Contract NAS 9-16639
Final Summary Report
30 April 1985

Prepared For:
NASA - Lyndon B. Johnson Space Center



Aerojet
TechSystems
Company

Final Summary Report 30 April 1985

IGNITION CHARACTERIZATION OF LOX/HYDROCARBON PROPELLANTS

Contract NAS 9-16639

Prepared For

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FOREWORD

This final summary report is provided in partial fulfillment of Contract NAS 9-16639, Ignition Characterization of LOX/Hydrocarbon Propellants. The program was conducted for the NASA - Johnson Space Center by the Aerojet TechSystems Company under the cognizance of W. C. Boyd, technical monitor. At Aerojet, the project engineers were B. R. Lawver and K. Y. Wong, and the program managers were R. W. Michel and D. C. Rousar. The technical work was performed during the period 6 July 1982 to 31 July 1984.

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ABSTRACT

This report summarizes the results of an evaluation of the ignition characteristics of the GOX/Ethanol propellant combination. Ignition characterization was accomplished through the analysis, design, fabrication and testing of a spark initiated torch igniter and prototype 620 lbF thruster/igniter assembly. The igniter was tested over a chamber pressure range of 3.3 to 262 psia and a mixture ratio range of 0.4 to 40. The prototype thruster/igniter assembly was tested over the chamber pressure range of 74 to 197 psia and mixture ratio range of 0.778 to 3.29. Cold (-92° to -165°F) and ambient (44° to 80°F) propellant temperatures were used.

Spark igniter ignition limits and thruster steady-state and pulse mode, performance. cooling and stability data are presented. Spark igniter ignition limits are presented in terms of cold flow pressure, ignition chamber diameter and mixture ratio. Thruster performance is presented in terms of vacuum specific impulse versus engine mixture ratio.

GOX/Ethanol propellants were shown to be ignitable over a wide range of mixture ratios. Cold propellants were shown to have a minor effect on igniter ignition limits. Thruster pulse mode capability was demonstrated with multiple pulses of 0.08 sec duration and less.

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I. INTRODUCTION

Recent studies indicate that future space transportation system (STS) engine development and operational recurring costs may be reduced through the use of low cost oxygen/hydrocarbon propellants in the auxiliary propulsion system (orbital maneuvering, reaction control, and vernier engines). The storable propellants (nitrogen tetroxide/monomethylhydrazine) used on the current STS are toxic and considerably more expensive than oxygen/hydrocarbon propellants. These propellants were selected over the oxygen/hydrocarbon propellants because a more extensive storable propellant thruster data base existed at that time. Also, the storable propellants are hypergolic and this eliminated the need for external ignition sources which were considered to be less reliable.

During the past five (5) years, several studies have been conducted for the purpose of expanding the oxygen/hydrocarbon thruster data base. Previous studies have examined the combustion performance and cooling characteristics of LOX/RP-1, LOX/propane, LOX/methane and LOX/ethanol. The objective of this study was to characterize the ignition and pulse mode capability of promising oxygen/hydrocarbon propellants for long-life, reusable, STS application. GOX/ethanol propellants were selected for testing on the basis of their demonstrated clean burning characteristics (Ref. 1) and because of the system advantages indicated for this propellant combination in the study reported in Reference 2.

In this program, a prototype 620 lbF GOX/ethanol RCE thruster with spark-initiated torch igniter was designed, fabricated and tested over a wide range of inlet conditions. The prototype igniter was used to determine the fundamental spark ignition characteristics of GOX/ethanol propellants prior to thruster testing. These ignition characteristics included the effects of igniter chamber diameter, cold-flow pressure level (inlet pressure), spark energy, and inlet temperature. An integral thruster/igniter assembly was used to evaluate thruster pulse mode capability with the GOX/ethanol propellant combination. The thruster was demonstrated to be ignitable over the whole range of inlet conditions. All of the tests demonstrate smooth combustion with no evidence of combustion instability. The exhaust plumes were clean with no evidence of soot or carbon deposition. Pulse mode capability was demonstrated with pulse durations down to 40 msec.

Two hundred five (205) igniter ignition tests were conducted covering a mixture ratio range of 0.4 to 40. Ignition limits were defined in terms of igniter mixture ratio, cold-flow pressure, spark energy and propellant temperature.

Eight-four (84) thruster tests were run covering a mixture ratio range of 0.78 to 3.29 and a chamber pressure range of 74 to 303 psia. Thruster pulse mode capability was demonstrated with pulse durations as short as 0.040 sec.

Two (2) additional igniter injectors were designed and fabricated for hydrocarbon fuel ignition testing at NASA/JSC with LOX/methane and LOX/propane propellants.

II. PROGRAM OBJECTIVES

The objectives of this program were to evaluate and characterize the ignition of promising low-cost liquid oxygen/hydrocarbon (LOX/HC) propellants for long-life reusable spacecraft propulsion systems. Basic ignition characteristics of the GOX/Ethanol propellant combination were generated and evaluated over a range of operating conditions applicable to an auxiliary propulsion system (orbital maneuvering, reaction control, and vernier engines) for the Space Shuttle Orbiter. The operational and design limitations associated with engine ignition for these propellants were identified and assessed. Information necessary for specific hardware designs was also generated.

The program consists of five parts; Task I - Igniter Experimental Evaluation; Task II - Preliminary Design; WBS 8.0 - Added Scope Testing; Task III - Thruster Experimental Evaluation; and WBS 9.0 - Added Scope Hardware Design and Fabrication.

The objectives of Task I were to determine the fundamental spark ignition characteristics of GOX/Ethanol propellants. These characteristics include the effects of design and operating variables such as igniter chamber size, cold-flow pressure level, spark energy, igniter cooling requirements, pulse mode capability and oxidizer manifold fuel contamination. A secondary objective was to evaluate the carbon formation potential of fuel rich gas generator mixtures. This was accomplished by designing a gas generator chamber with an internal secondary injector to fit the igniter injector. Nine (9) tests were run over a mixture ratio range of 0.189 to 0.441. No evidence of carbon formation was observed. Detailed results are reported in the Task I data dump, Reference 3.

The objective of Task II was to analytically evaluate the ignition, performance and cooling requirements of GOX/Ethanol igniters and thrusters for auxiliary propulsion applications. The results of these evaluations were used to guide the Task I and III igniter and thruster design activities. These results are reported in the Task II data dump, Reference 4.

The objective of the added scope testing was to get an early evaluation of combustion performance of GOX/Ethanol thrusters for auxiliary propulsion applications. These tests were conducted using residual hardware from the mid Pc (NAS 9-15958) program. The results of these evaluations were an aid to the Task III thruster design activity. Complete details are presented in the WBS 8.0 data dump, Reference 5.

The primary objective of the Task III testing was to experimentally evaluate GOX/Ethanol thruster ignition and pulse-mode operation. A secondary objective was to evaluate thruster steady-state performance, cooling and stability characteristics. A prototype thruster consisting of an igniter, injector, and thrust chamber was tested over a wide range of propellant inlet temperatures and pressures.

The testing was conducted in two parts. The first part checked out the Task I GOX/Ethanol igniter in the thruster test facility. Nineteen (19) tests were run to evaluate the igniter feed system, valve timing, and propellant inlet condition effects.

The second part addressed full thruster operation. Sixty-five (65) thruster tests were conducted. The first thirteen (13) tests were run with a heat sink chamber to evaluate ignition and inlet condition effects on performance. The remaining tests were run with a thin-wall chamber to evaluate film-coolant effectiveness and pulse mode performance over a range of inlet conditions. Detailed data are presented in the Task III data dump, Reference 6.

II, Program Objectives (cont.)

The objective of the added scope design and fabrication tests was to design and fabricate two (2) additional igniter injectors for ignition testing with LOX/Methane and LOX/Propane propellants. The ignition testing will be conducted at NASA/JSC.

III. METHOD OF APPROACH

The prototype 620 lbF GOX/Ethanol RCS thruster with integral spark-initiated torch igniter, shown in Figures 1 and 2, was designed, fabricated and tested over a wide range of inlet conditions. The prototype igniter shown in Figures 3 and 4 was used to determine the fundamental spark ignition characteristics of GOX/Ethanol propellants prior to thruster testing. These ignition characteristics included the effects of igniter chamber diameter, cold-flow pressure level (inlet pressure), spark energy, and inlet temperature. Two hundred five (205) igniter hot fire tests were run over a mixture ratio range of 0.4 to 40 and a cold-flow pressure range of 3.3 to 49 psia. Chamber diameters of 0.15, 0.20 and 0.30 were tested. Propellant temperatures ranged from 80° to -165°F.

The ignition tests were conducted using the start sequence shown in Figure 5. The oxygen flow was started 10 msec ahead of the Ethanol flow. The spark discharge was delayed about 110 msec to establish the cold-flow pressure. Ignition or non-ignition was indicated by the chamber pressure and exhaust temperature response. The total test duration was 0.4 seconds.

The gaseous oxygen inlet pressure was set to achieve the desired chamber cold-flow pressure and the fuel inlet pressure varied up or down to shift the mixture ratio. The spark energy was varied at fixed inlet conditions to determine its effect on ignition limits.

The ignition test variables and the range of conditions tested are listed in Table I. The effects of these variables were determined by plotting the ignition response (i.e., ignition or non-ignition) on the flame quench parameter (PD) versus igniter core mixture ratio coordinates. The flame quench parameter is the product of the cold-flow pressure before ignition and the chamber diameter.

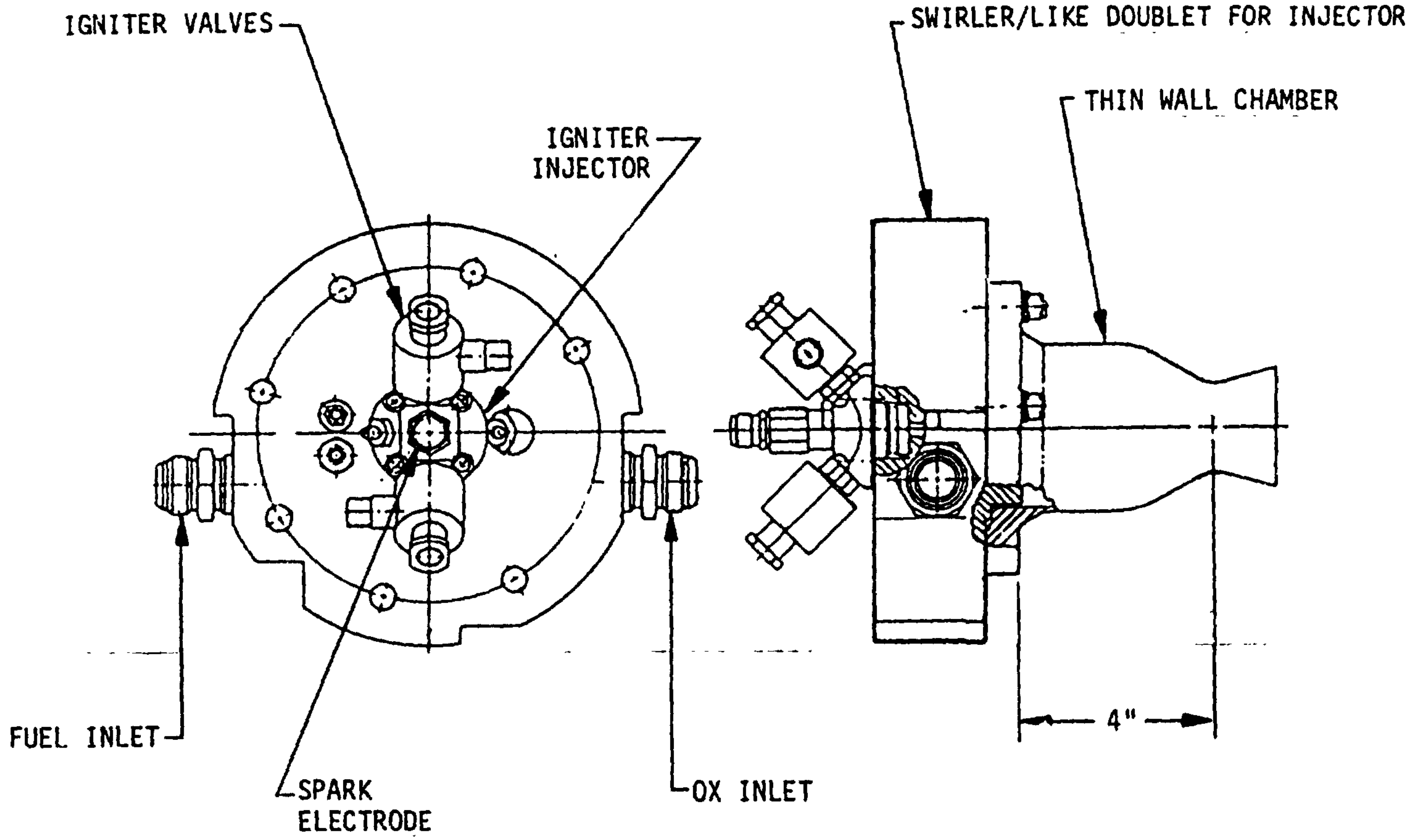

Figure 1. GOX/Ethanol Thruster

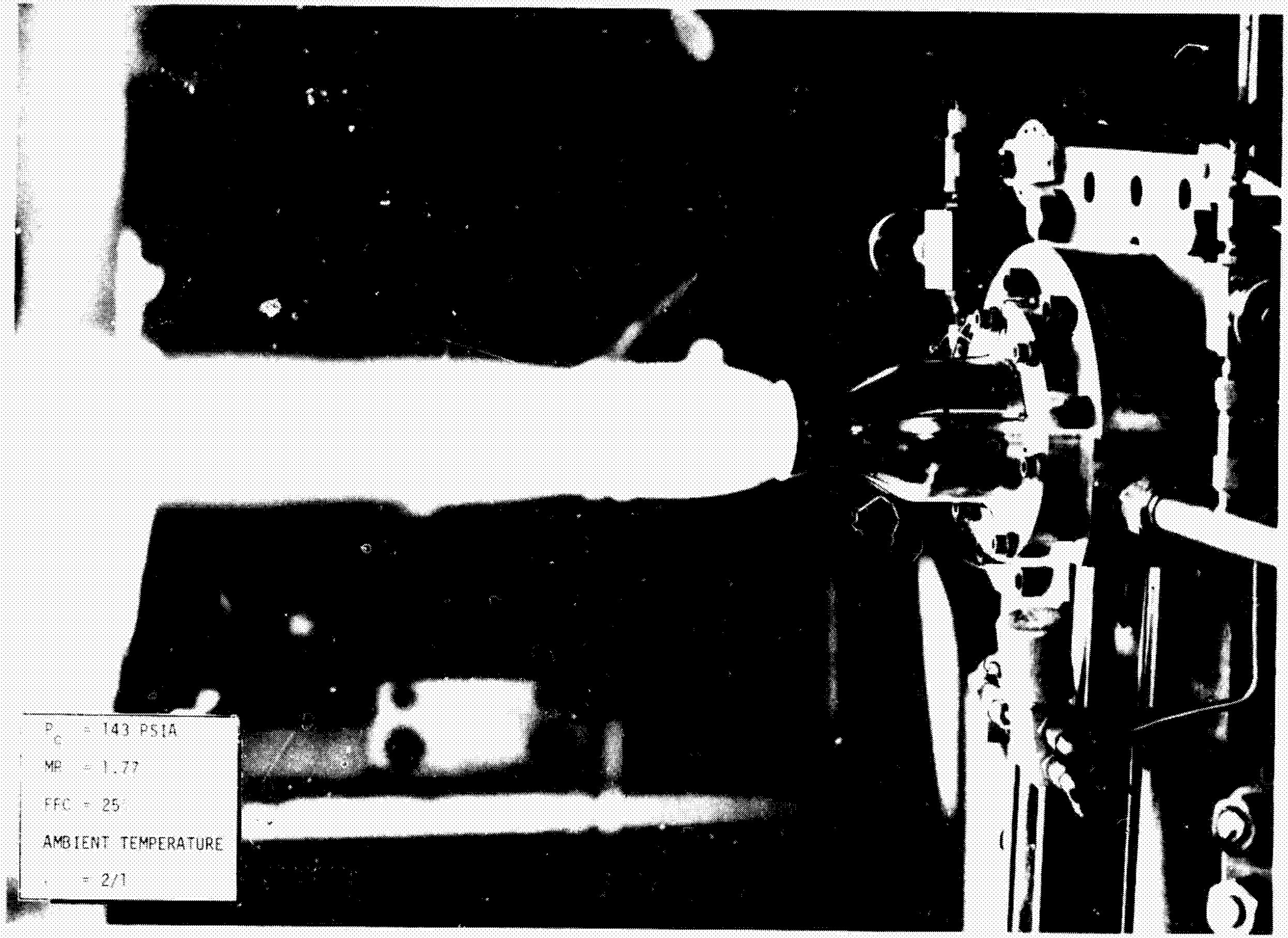

Figure 2. Task I Ignition Test Sequence

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Igniter Test Assembly

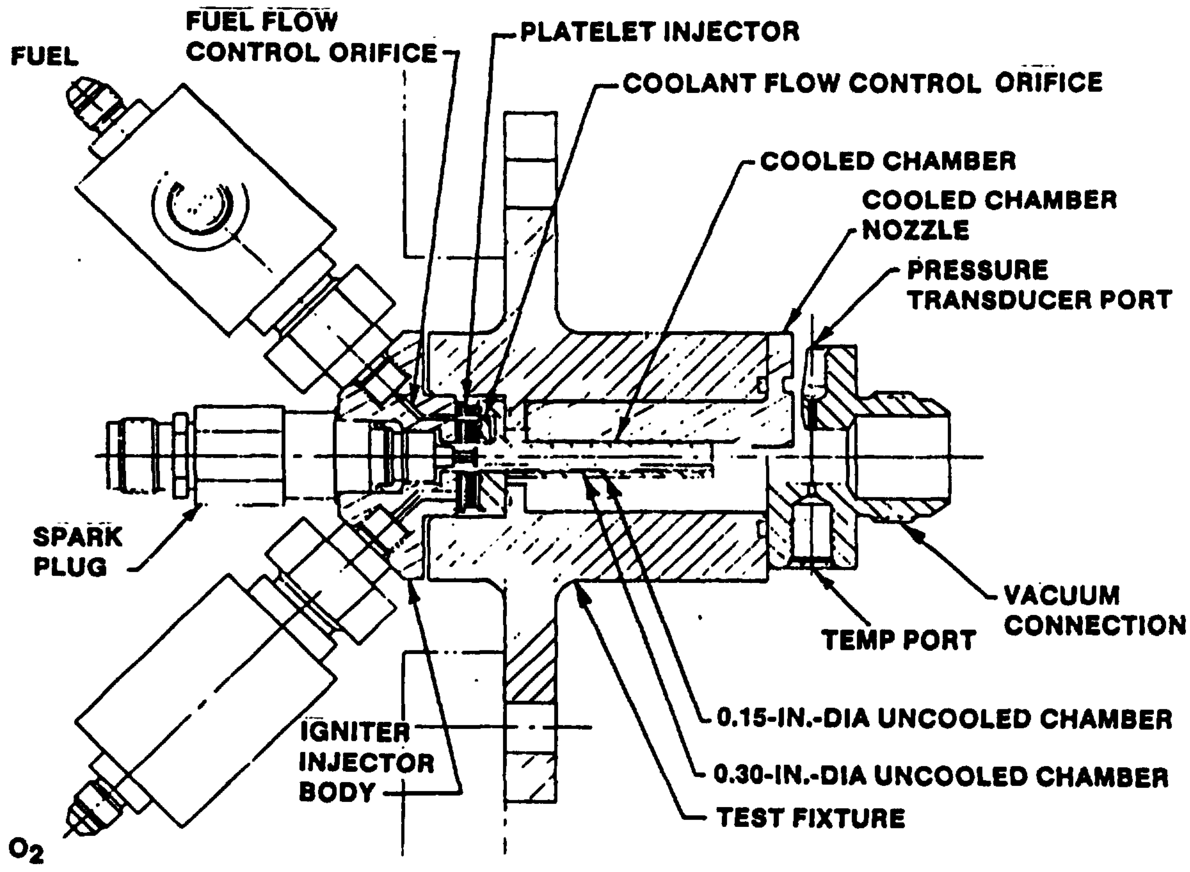

Figure 3. Task I Igniter and Test Hardware Assembly

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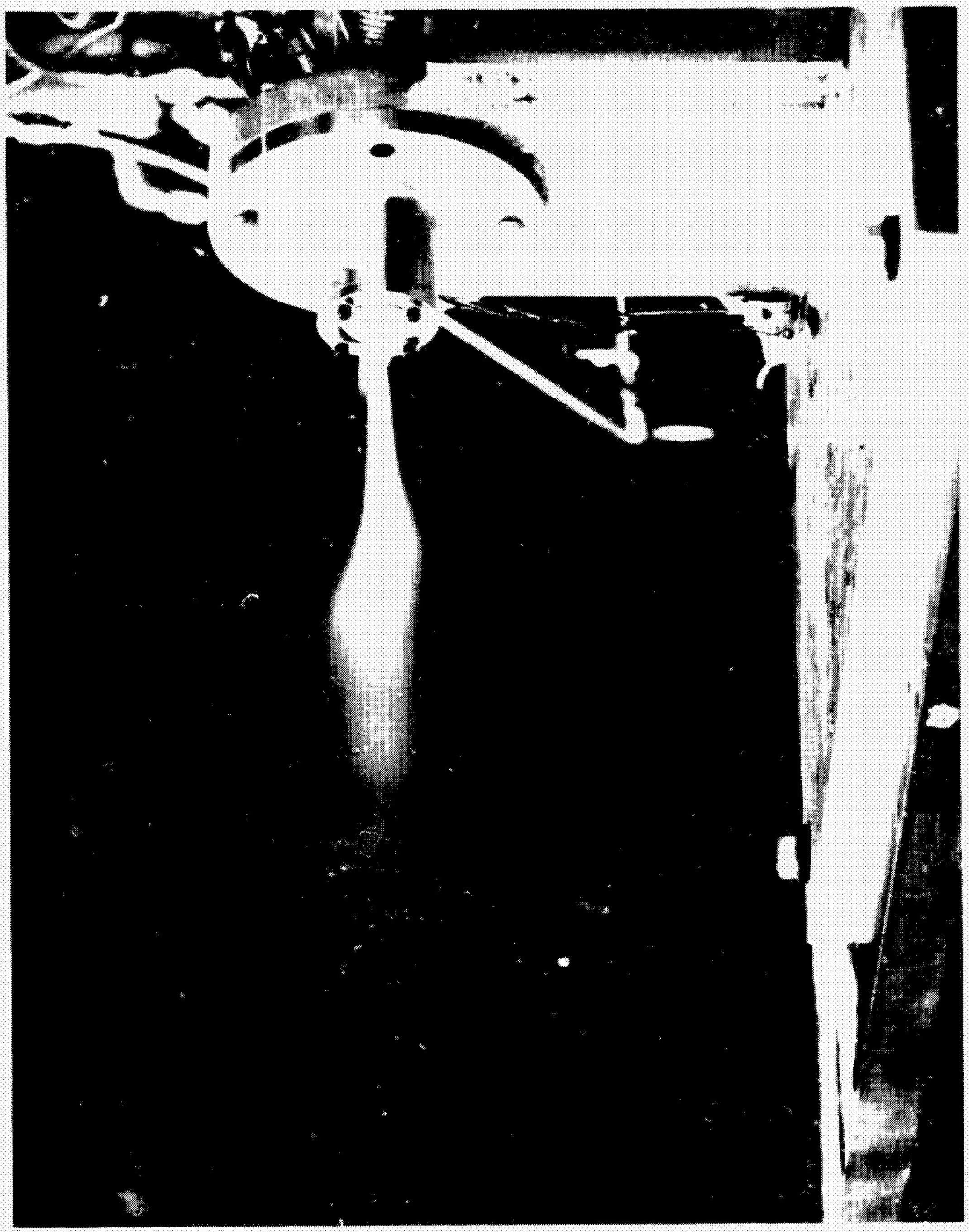

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Figure 4. GOX/Ethanol Igniter Test 281

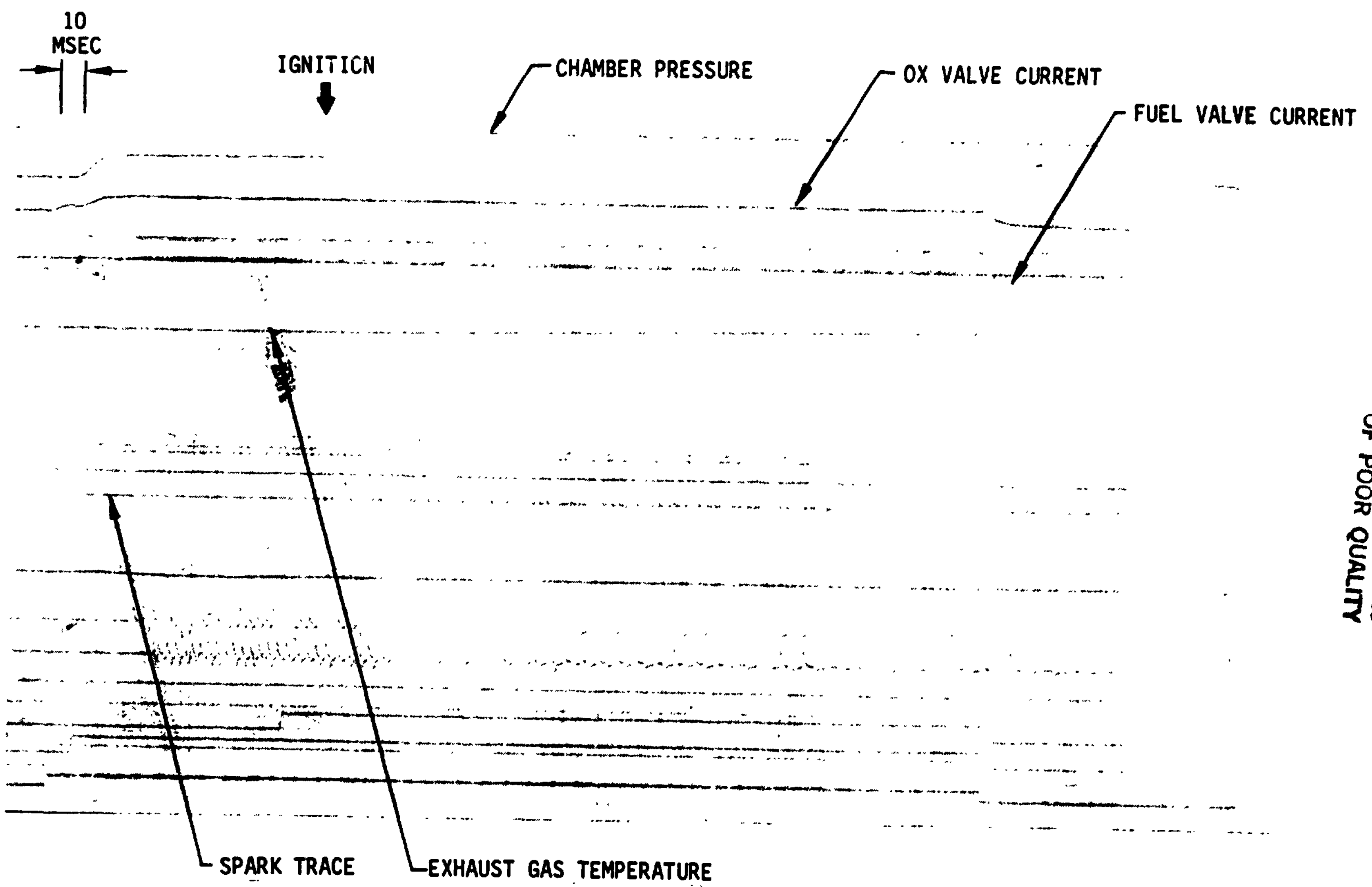

Figure 5. Task I Ignition Test Sequence

TABLE I

TASK I IGNITER TEST VARIABLES

- Propellant Temperature
  - Amb 44 to 80°F
  - Cold -92 to -165°F
- Cold Flow Pressure
  - 3.3 - 49 psia
- Chamber Diameter
  - 0.15 in.
  - 0.30 in.
- Mixture Ratio
  - 0.4 - 40
- Spark Energy
  - 10 - 50 mJ
- Spark Rate
  - 300 SPS (Fixed)
- Spark Gap
  - 0.025 in. (Fixed)

The integral thruster/igniter assembly was used to evaluate thruster pulse mode capability with the GOX/Ethanol propellant combination. Thirteen heat sink chamber and 52 thin wall chamber tests were run over the mixture ratio range of 0.778 to 3.29 and chamber pressure range of 74 to 197 psia. Propellant temperatures ranged from ambient to -165°F.

The thruster assembly was instrumented to measure thrust, propellant flow-rates, inlet pressures and temperatures, combustor wall temperatures, and combustion pressures. These data were recorded on both digital and analog recording instruments.

Thruster tests were run to evaluate ignition, performance, thermal compatibility, stability and pulse mode operation. The primary test variables were propellant inlet pressures and temperature. Secondary variables were film coolant injection scheme (tangential or swirl) and valve actuation gas (GN_2 or GHe). Most of the tests were run with ambient temperature propellants and the thin-wall chamber.

An oscillograph trace of a typical thruster start sequence for a steady-state firing is shown in Figure 6. The igniter fuel valve was signaled 10-20 milliseconds ahead of the igniter oxidizer valve to provide a momentary core fuel rich condition. The igniter-alone testing showed this to be a more reliable start condition than simultaneous igniter valve actuation. The spark was signaled on with the fuel valve so that ignition would occur immediately upon introduction of the oxidizer. The thruster bipropellant valve was signaled on about 45 milliseconds from the igniter fuel valve signal to permit time for the computer to test the igniter chamber pressure. If the igniter chamber pressure had not achieved a preselected level, then the test would have been terminated. Ignition was achieved on all of these tests.

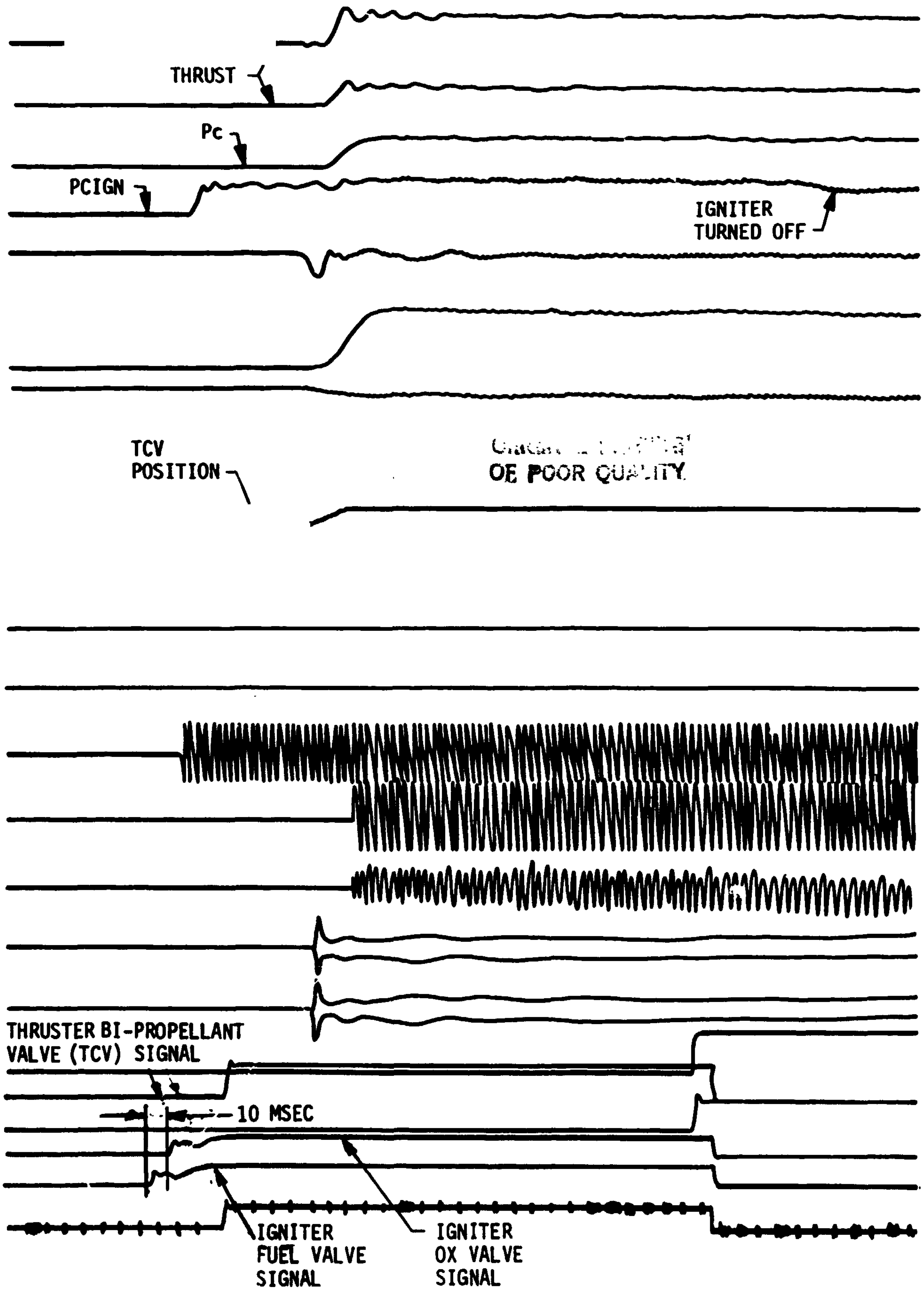

Figure 6. Thruster Ignition Sequence - Steady State Testing

An oscillograph trace of a typical thruster pulse sequence is shown in Figure 7. The pulse test variables were valve actuation (GN_2 or GHe), pulse width (valve open to valve close), inlet pressures and inlet temperature (ambient or cold). The first four (4) pulse tests were run with a GN_2 driven actuator. The remaining tests used a GHe driven actuator to speed the valve opening and closing times.

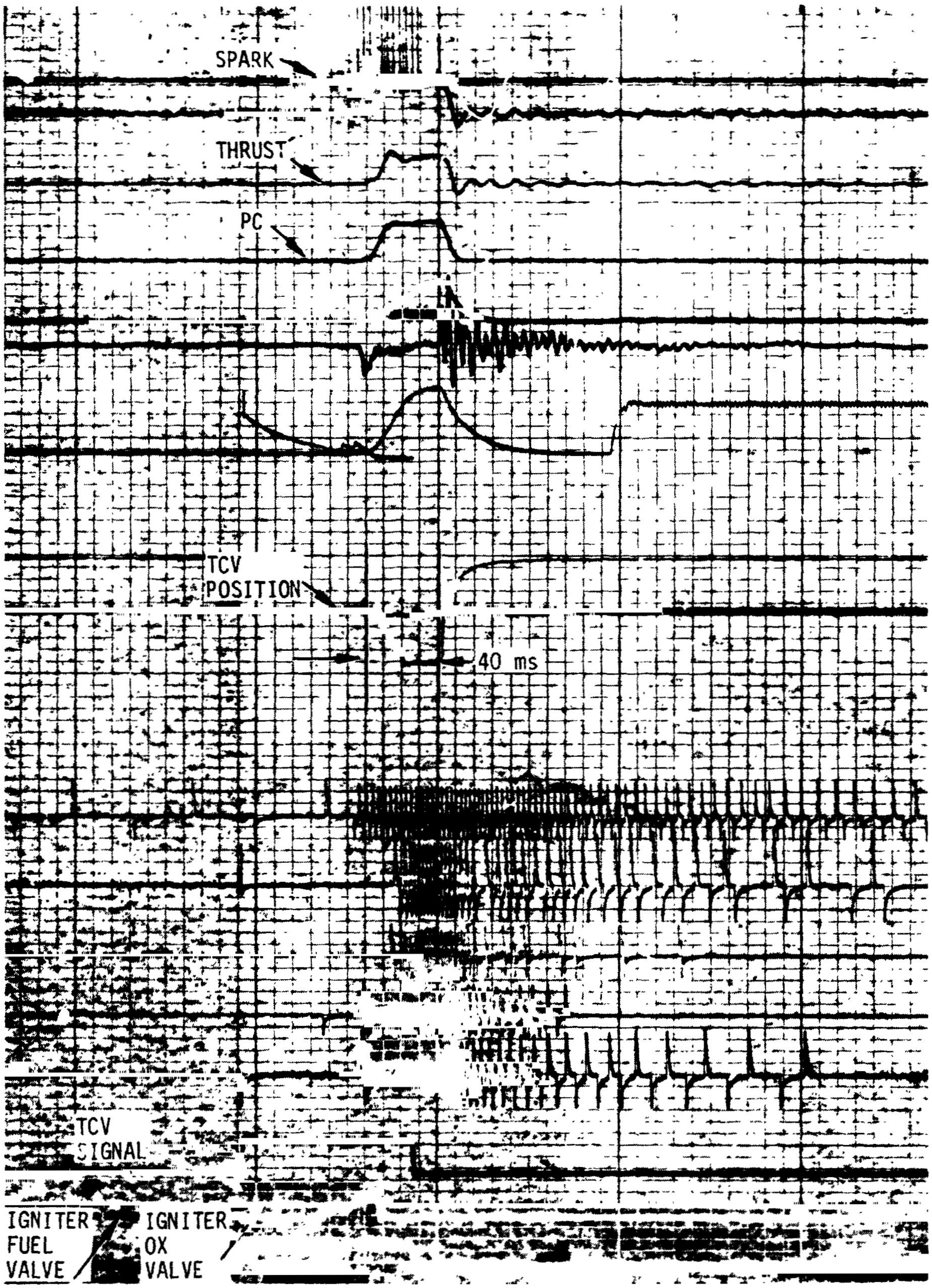

Figure 7. Thruster Pulse Sequence - GHe Actuator

IV. SIGNIFICANT RESULTS

The ignition and steady-state performance characteristics of spark ignited GOX/Ethanol propellants were defined and pulse mode capability was demonstrated. The prototype igniter was tested over the range of conditions listed in Table II. A total of one hundred thirty-eight (138) igniter tests were run to define the ignition limits in terms of cold flow pressure, ignition chamber diameter, mixture ratio, spark energy and propellant temperature.

The effect of cold flow pressure (P) and chamber diameter (D) on the flame quench ignition limits were correlated as shown in Figure 8. Cold (-125°F) fuel was found to only slightly reduce the ignition limits. Cold oxidizer had no measurable effect on ignition. Reducing the spark energy from 50 MJ/spk to 10 MJ/spk has a more significant effect on the ignition limit as indicated in Figure 9.

Sixty-seven additional igniter tests were run to verify igniter cooling, C* performance and pulse mode capability. Ignition was achieved at all conditions where ignition was predicted.

A total of eighty-four thruster tests were made. Nineteen added scope tests were made using residual hardware (Contract NAS 9-15958) and sixty-five tests were made using the prototype thruster (Figure 1) designed and fabricated on this program. The results of the added scope tests were used to confirm the performance of a swirler/like doublet OFO injector for the prototype thruster.

The previous thruster test data are summarized in Table III. Combustion and Isp efficiencies are shown in Figure 10 for ambient and cold propellant. The ambient temperature performance is nearly flat over the whole range of mixture ratio tested. The cold propellant reduces Isp efficiency by about 6% at the design point mixture ratio. Twenty five percent (25%) fuel film collant was used on all of the prototype thruster testing.

TABLE II

TASK I GOX/ETHANOL IGNITION AND DURABILITY TEST MATRIX

		Tests							Test Conditions		
Status		Series	# of Tests Actual	# of Tests Planned	Feed/Temp Ox Fuel	Chamber Diameter	Core MR	Film Cooling	Duration	Chamber Pressure	Remarks
Complete	Checkout	A	6	2	Amb/Amb	D-1	TBD	None	0.2	TBD	Checkout
Complete		B	32	25	Amb/Amb	D-1	1-40	None	0.2	1-50	Baseline ignition with ambient propellant
Complete		C	21	4	Amb/Amb	D-1	TBD	None	0.2	TBD	Increased spark at no light conditions
		C-1	4	-	Amb/Amb	D-1	TBD	None	1.0	75-150	Determine heat flux
Complete		D	17	12	Cold/Cold	D-1	TBD	None	0.2	1-50	Ignition with cold propellants
Complete	Ignition Tests	E	4	4	Cold/Cold	D-1	TBD	None	0.2	TBD	Increased spark at no light condition
Complete		F	7	6	Amb/Cold	D-1	TBD	None	0.2	TBD	Cold fuel sensitivity
Complete		G	11	6	Cold/Amb	D-1	TBD	None	0.2	TBD	Cold GOX sensitivity
Complete		H	22	10	Amb/Amb	D-2	1-40	None	0.2	1-50	Reduced dia. sensitivity; ambient propellant
Complete		I	14	10	Cold/Cold	D-2	1-40	None	0.2	1-50	Reduced dia. sensitivity; cold propellant
Complete	Cooling Tests	J	3	2	Amb/Amb	D-4	TBD	TBD	0.2	TBD	Checkout with film cooling
Complete		K	9	6	Amb/Amb	D-4	1-40	Nom	TBD	1-50	Verify ignition characterization with cooling; ambient propellant
Complete		L	9	6	Cold/Cold	D-4	1-40	Nom	TBD	1-50	Verify ignition characterization with cooling; cold propellant
Complete		M	6	5	Amb/Amb	D-4	3-20	TBD	5	150	Film cooling effectiveness vs core MR
Complete		N	2	2	Amb/Amb	D-4	3-20	TBD	5	75	Low Pc sensitivity
Complete		O	5	2	Amb/Amb	D-3	TBD	TBD	5	TBD	Low MR coking at gas generator condition
Complete		P	9	5	Cold/Cold	D-4	3-20	TBD	5	150	Film cooling effectiveness at low propellant temperature

TABLE II (cont.)

Status		Tests: Series	# of Tests: Actual	# of Tests: Planned	Feed/Temp Ox Fuel	Test Conditions: Chamber Diameter	Core MR	Film Cooling	Duration	Chamber Pressure	Remarks
Complete	Cooling Tests	O	4	2	Cold/Cold	D-3	TBD	TBD	5	TBD	Low MR coking at low-temperature gas generator conditions
Complete		R	4	3	Amb/Amb	D-4	TBD	TBD	5	150	Film-cooling optimization
Complete		S	3	3	Cold/Cold	D-4	TBD	TBD	5	150	Film-cooling optimization
Complete		T	3	3	Amb/Amb	D-4	TBD	TBD	0.1	150	Igniter pulse and restart with ambient propellant (2 firings per test)
Complete		U	3	3	Amb/Amb	D-4	TBD	TBD	0.1	150	Igniter pulse and multiple restart.
Complete		V	8	3	Cold/Cold	D-4	TBD	TBD	0.1	150	Igniter pulse and restart with cold propellants
Complete		W	10	15	TBD	TBD	TBD	TBD	TBD	TBD	Contingency tests
			216	140							

NOTE:
1. Adequate altitude conditions maintained through ignition period and through pulse, coast, and restart period.
2. Chamber pressure refers to cold flow pressure for series A through I and to steady-state pressure for series J through W.
3. Cold propellant temperatures are approximately 50°F above the fuel freezing temperature.
4. D-1 = 0.3" diameter;
   D-2 = 0.15" diameter;
   D-3 = Gas Generator Chamber;
   D-4 = 0.2" diameter

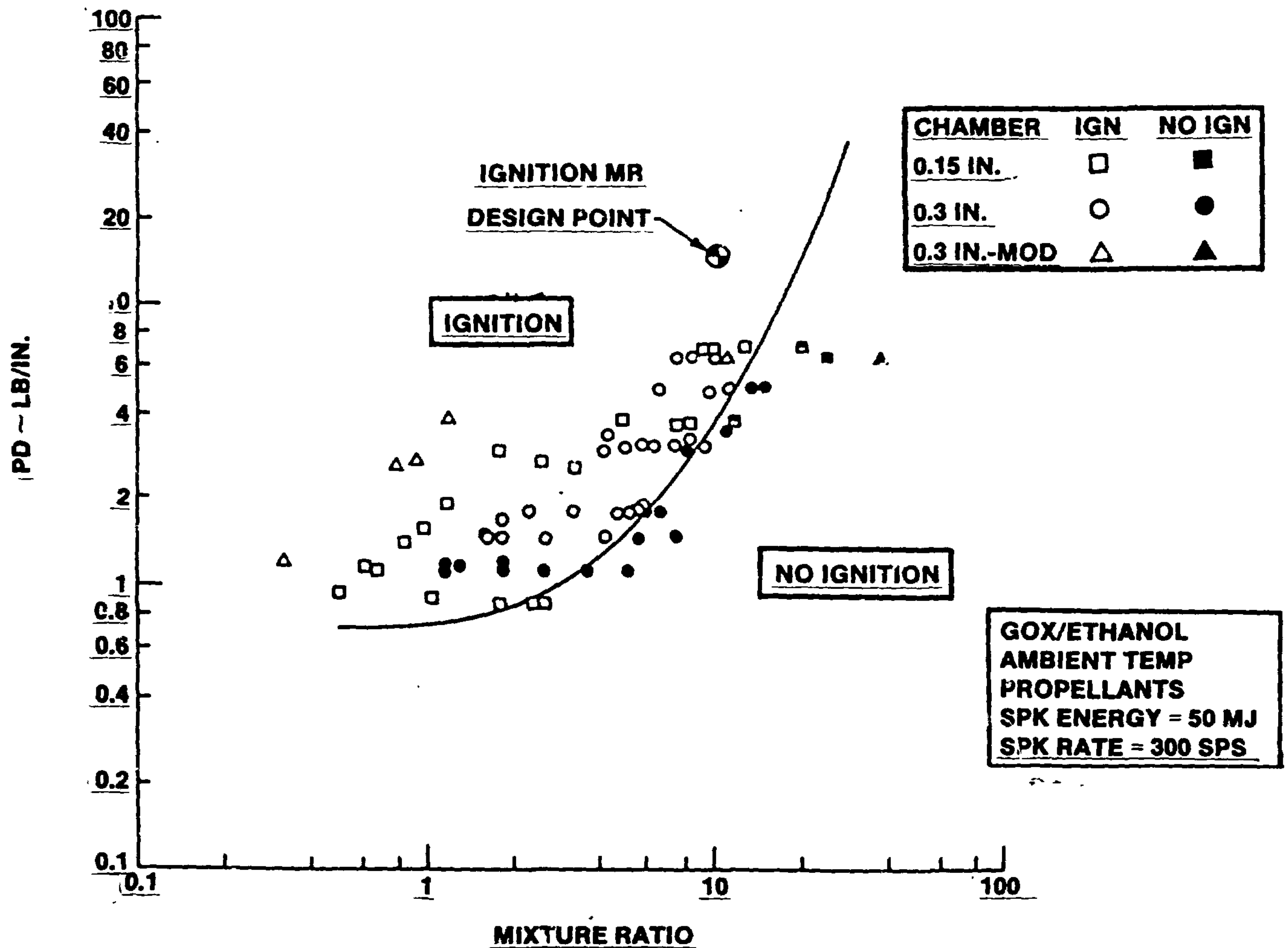

Figure 8. Effect of Chamber Diameter and Cold-Flow Pressure on Ignition

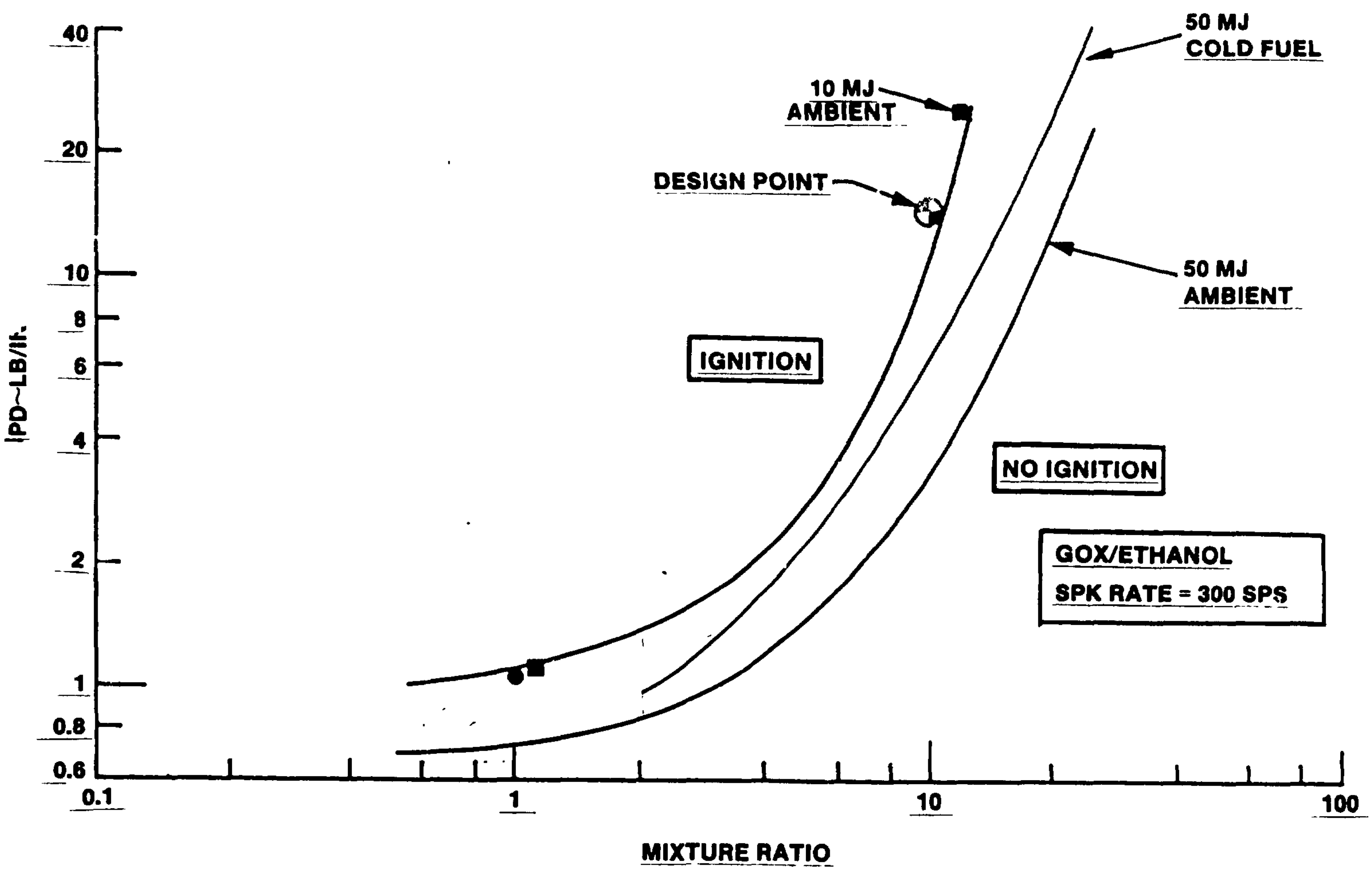

Figure 9. Effect of Spark Energy and Cold Fuel on Ignition

TABLE III

PROTOTYPE THRUSTER TEST DATA SUMMARY

Test Objective	Propellant Temperature	Chamber	Valve Actuation	Film Coolant	MR Range	Pc Range (psia)	Test Duration (sec)	No. of Pulses	Coast Duration (sec	No. of Tests	No. of Igniter Ignitions	No. o Thruste Ignitic
Ignition and Performance	Ambient	Heat Sink	GN_2	Tangential	0.79-2.5	92-190	0.5-2	N/A	N/A	13	13	13
Performance and Cooling	Ambient	Thin-Wall	GN_2	Tangential	0.78-2.71	91-191	1-5	N/A	N/A	11	11	11
Pulse Capability	Ambient	Thin-Wall	GN_2	Tangential	1.32-1.36	150-160	0.18-0.475	4	N/A	4	4	4
Performance and Cooling	Ambient	Thin-Wall	GHe	Swirl	0.89-2.86	91-197	1-5	N/A	N/A	13	12	12
Pulse Capability	Ambient	Thin-Wall	GHe	Swirl	0.94-2.80	94-195	.040-.080	47	0.08-1.0	11	41	41
Performance and Cooling	Cold	Thin-Wall	GHe	Swirl	1.24-3.29	74-188	1-5	N/A	N/A	8	8	8*
Pulse Capability	Cold	Thin-Wall	GHe	Swirl	0.81-1.37	84-172	.040-.080	35	0.08	5	28	28
										65	118	118

Nominal Pc = 150 psia

Nominal MR = 1.8

Nominal FFC = 25%

Nominal Igniter MR = 1.0

SPK Energy = 50 MJ/SPK

SPK Rate = 300 SPS

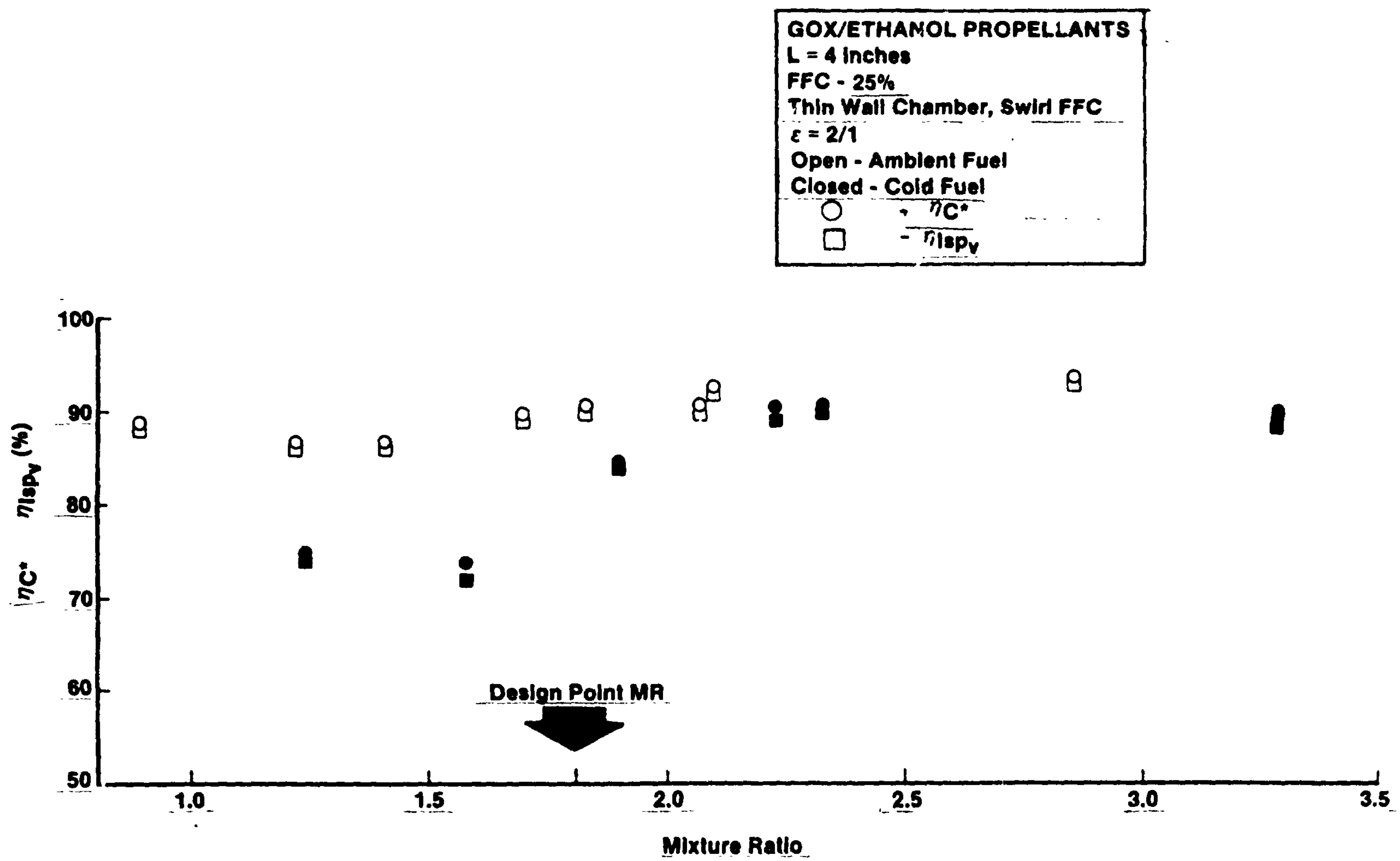

Figure 10. Thruster C* and Isp_V Efficiency

The thin-wall chamber thermocouple data with 25% FFC indicate that 40% FFC will be required to cool a columbium radiation cooled flight engine. Additional testing will be required to define performance for 40% FFC.

Thruster pulse performance was determined over a pulse width range of 40 msec to 500 msec. The Bit Isp is shown in Figure 11.

All of the thruster tests were smooth with no evidence of combustion instability. The exhaust plumes were observed to be clear and clean. No evidence of carbon formation or deposition was found.

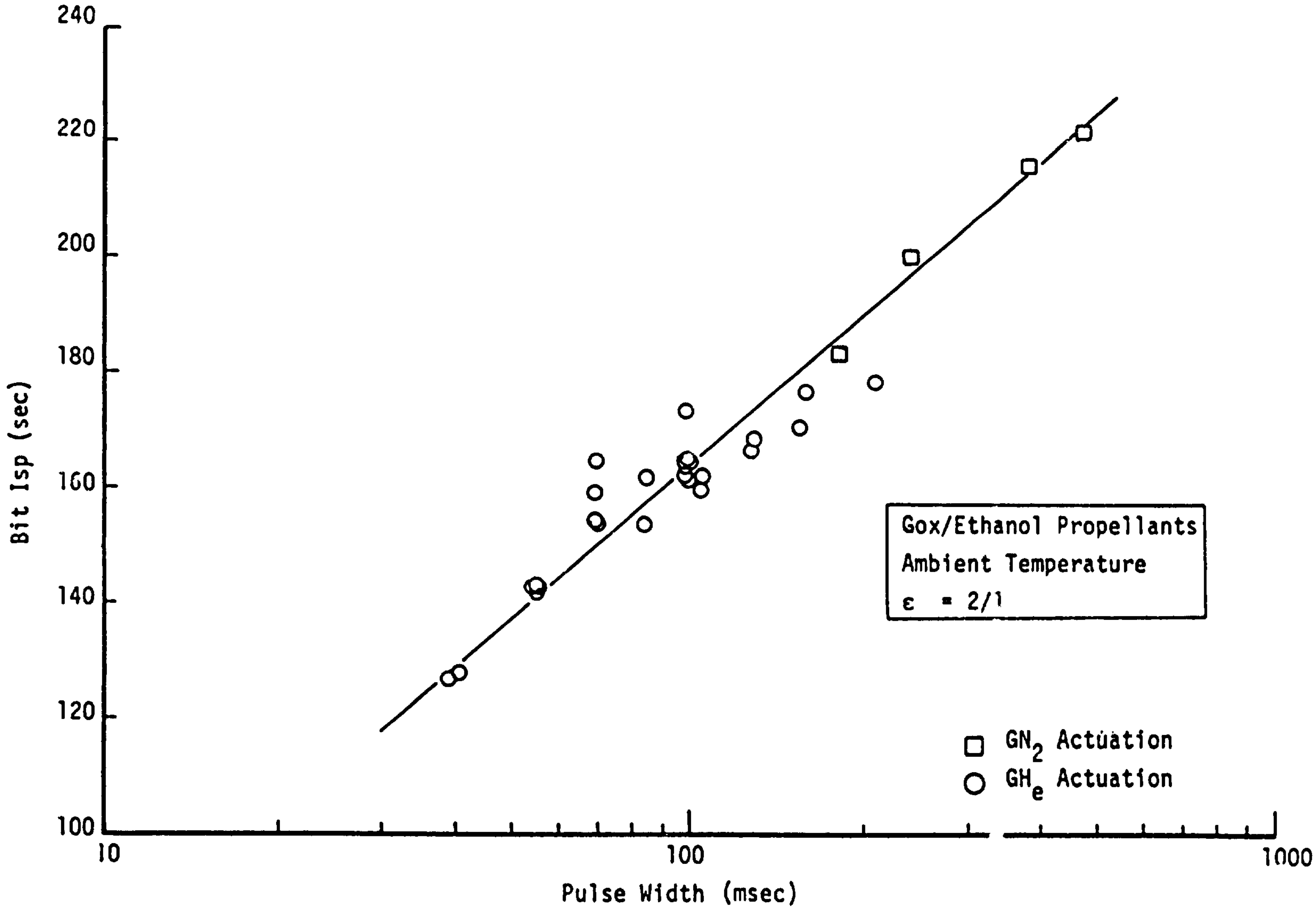

Figure 11. Thruster Bit Isp

V. LIMITATIONS

The applicability of the GOX/Ethanol thruster combustion data obtained on this program are limited to the 0-300 psia pressure range and the 0.778 to 3.29 mixture ratio range. The spark ignition data are applicable to the 0.4 to 40 mixture ratio range. Propellant temperatures ranged from ambient down to -156°F. The fuel film coolant data are limited to 26% FFC. No other data limitations are known.

VI. RECOMMENDED FOLLOW-ON WORK

The following additional work is recommended.

1. Further GOX/Ethanol thruster testing should be conducted to evaluate axial fuel film coolant injection and performance using up to 40% FFC and columbium chambers to finalize FFC requirements for RCS thrusters.

2. Extreme fuel rich (MR <0.1) GOX/Ethanol igniter testing should be conducted to completely define the fuel-rich ignition limit.

3. Ignition testing with other hydrocarbon fuel (i.e., Propane and Methane) should be done to expand the hydrocarbon fuel data base.

VII. CONCLUSIONS

It is concluded that spark ignition of the GOX/Ethanol propellant combination is reliable and that thruster pulse mode capability was demonstrated. The spark ignition limits were defined and a GOX/Ethanol thruster data base generated.

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