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ALTITUDE IGNITION STUDIES

C. R. MASTROMONICO, 1ST LT USAF

TECHNICAL REPORT AFRPL-TR-71-70

JULY 1971

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13. ABSTRACT

A facility for analyzing the effects of engine hardware parameters on the pulse characteristics of bipropellant attitude control thrusters in a space environment was constructed at the Air Force Rocket Propulsion Laboratory (AFRPL). In a series of tests involving 32 engine configurations, 17 engine parameters were varied. The propellants considered were N_2H_4-UDMH (50/50) and MMH fuels and NTO as the oxidizer. The experimental phase of the program was based on a statistical test matrix designed to minimize the number of firings and the hardware requirements. This report includes a detailed description of the altitude ignition facility and presents an analysis of the system and the data obtained. Data obtained from the altitude ignition facility was reviewed and found to be unacceptable for use in analyzing the effects of engine hardware variations on the pulse characteristics of an attitude control thruster. This was because propellant valve responses were not repeatable within the limit of error necessary to obtain meaningful transient pressure data. The source of the error in valve response was found to be irregularities in the internal parts of the propellant valves. Since the problem could not be resolved in time to meet the needs of associated AFRPL programs, the Altitude Ignition Program has been discontinued. The data indicated that the ignition delay times for the fuels studied compared as follows: $N_2H_4 > MMH \approx 50/50 > UDMH$. It also showed that increasing oxidizer lead shortened ignition delay times and amplified pressure spiking.

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FOREWORD

This report covers work on Project 314803 DRV, "Altitude Ignition Studies," by the Engine Components Branch of the Liquid Rocket Division of the Air Force Rocket Propulsion Laboratory (from July 1968 to December 1970).

The project engineers were Jack E. Hewes for the design phase, Capt Alan W. McPeak for the buildup and experimental phases, and 1st Lt Charles R. Mastromonico for the data and system analysis phase of the program.

This technical report has been reviewed and is approved.

EDWARD E. STEIN, Chief
Propulsion Subsystems Branch
Liquid Rocket Division
Air Force Rocket Propulsion Laboratory

ABSTRACT

A facility for analyzing the effects of engine hardware parameters on the pulse characteristics of bipropellant attitude control thrusters in a space environment was constructed at the Air Force Rocket Propulsion Laboratory (AFRPL). In a series of tests involving 32 engine configurations, 17 engine paramenters were varied. The propellants considered were N_2H_4-UDMH (50/50) and MMH fuels and NTO as the oxidizer. The experimental phase of the program was based on a statistical test matrix designed to minimize the number of firings and the hardware requirements. This report includes a detailed description of the altitude ignition facility and presents an analysis of the system and the data obtained.

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The data indicated that the ignition delay times for the fuels studied compared as follows: $N_2H_4 > MMH \approx 50/50 > UDMH$. It also showed that increasing oxidizer lead shortened ignition delay times and amplified pressure spiking.

TABLE OF CONTENTS

ILLUSTRATIONS

LIST OF TABLES

SECTION I

INTRODUCTION

Ignition delays leading to the formation of large pressure spikes have been observed during the pulse-mode operation of bipropellant attitude control thrusters. At simulated high altitudes, the pressure spikes observed were severe enough to damage the thrust chambers and injector surfaces, and the ignition delays were of length significant enough to affect repeatable performance.

Thiokol/RMD investigated the processes occurring during pre-ignition chamber pressurization and compiled a theoretical computer model, the altitude ignition model, which adequately predicted the ignition delay and the pre-ignition chamber pressure history of two 50-pound thrusters using nitrogen tetroxide (NTO) as an oxidizer and hydrazine-type fuels. The in-house Altitude Ignition Program was initiated at the Air Force Rocket Propulsion Laboratory (AFRPL) in 1967 for the purpose of providing data to determine the influence of selected hardware parameters on chamber pressurization and ignition delay, and to incorporate this information into the Thiokol computer model. In addition, the data was to be used to verify the reliability of an overall pulse-mode performance model presently being developed under Contract AF04(611)-70-C-0074 by Rocketdyne.

SECTION II

BACKGROUND

In the early 1960's NASA switched from compressed gas to bipropellant attitude control thrusters for space applications. Thrusters using a propellant combination of NTO and hydrazine-type fuels experienced unexplained pressure spiking and ignition delays during pulse-mode operation at space ambient conditions. The pressure spiking severely damaged the combustion chambers and injector surfaces, and the unpredictable ignition delays limited thruster performance.

In July 1964, AFRPL authorized Thiokol/RMD to study the causes of these phenomena under contract AF04(611)-9946 and to compile a computer model to predict their occurrence (Reference 1). A follow-on contract (AF04(611)-11630) was granted to Thiokol by AFRPL in June 1966 to refine this computer model by including the effects of injector flashing, pre-ignition wall heat transfer, and the formation of a pre-ignition reaction to adduct (Reference 2). With these refinements, the model adequately predicted the pre-ignition chamber pressurization, the ignition delay, and the buildup of reaction adduct in a 50-pound thruster using the NTO/MMH propellant combination. In February 1968, a second follow-on contract (AF04(611)-0511) was granted to Thiokol for the purpose of generalizing the computer model and simplifying the input parameters prior to incorporating them into the overall pulse-mode performance model (Project ACE) (Reference 3). The result was a workable fundamental computer model which satisfactorily predicted the pre-ignition chamber pressure history of three propellant combinations in two different engine configurations.

In July 1967, the Altitude Ignition Facility was designed at AFRPL to provide data so that the effects of engine hardware parameters could be analyzed and included in the Thiokol altitude ignition model.

SECTION III

APPROACH

DATA TEST MATRIX DEVELOPMENT

From the results of the Altitude Ignition Model (Reference 3), Thiokol/RMD and AFRPL compiled a list of 17 engine parameters which were thought to have an effect upon the pulse characteristics (Table I).* These parameters were varied according to a statistical test matrix designed to yield the maximum amount of information using a minimum of hardware and run time. The test matrix shown in Figure 1 has 17 columns and 32 rows corresponding to the number of parameters included and the minimum number of engine configurations required to complete the statistical data requirement. The in-house facility was constructed at AFRPL according to the design of a facility located at Thiokol, and the parameter levels listed in Table I are based on this design. The plus (+) and minus (-) signs refer to the two different parameter levels which may be used for a particular configuration.

GENERAL PULSE DESCRIPTION

A single pulse from an attitude control thruster may be divided into three main time periods: (1) the start transient, (2) the steady-state period, and (3) the decay transient.(See Figure 2.) The start and decay transients are approximately reproduced for each pulse in a duty cycle. The steady-state period may differ from pulse to pulse depending on the pulse duration. For short-duration pulses, the transient periods represent a major contribution to the total pulse and an accurate evaluation of all three periods is necessary for a complete duty cycle evaluation. The transient periods illustrated in Figure 2 are divided further into several

*Figures and tables are presented sequentially beginning on pages 12 and 17, respectively.

time periods marking the start and completion of significant events such as: (1) the filling of the injector manifold, (2) the pre-ignition pressurization of the thrust chamber, (3) the blowdown in the chamber after the primary combustion processes have ceased, and (4) the boil-off of propellant in the hot manifold after the propellant valves have closed.

The pulse duration of an attitude control thruster is often as short as 200 milliseconds. During this time, the thruster chamber pressure may rise as high as 300 psi and the temperature above 1000°F. The events occurring between valve activation and ignition (i.e., from on-signal to the completion of period (B)) are described by the Thiokol altitude ignition model. The time period involved may be less than 30 milliseconds.

The pulse illustrated in Figure 2 does not reflect the effects of ignition delay and pressure spiking. Ignition delays of 10 to 20 milliseconds have been observed by Thiokol. Their effect on the pressure pulse is to extend the pre-ignition pressurization time (period B in Figure 2). For short-duration pulses, this effect may result in not attaining steady-state at all, yielding less than expected. On the other hand, pressure spiking to pressures three to four times greater than the normal level of overshoot, illustrated in Figure 2, results in a greater thrust than expected.

To obtain experimental data which accurately represents the processes occurring during a pressure pulse, the timing of the propellant injection system must be controlled precisely and the response of the instrumentation must be quick enough to follow the rapid rate of change of the system variables.

APPARATUS

The in-house Altitude Ignition Facility (Figures 3 and 4) was designed to provide a simulated space environment into which a 50-pound bipropellant thruster could be fired in a pulse mode and its pulse characteristics analyzed. The apparatus is divided into three distinct systems. The vacuum system is made up of everything to the left of the heavy dashed line in Figure 4. It consists of the following: a vacuum chamber, two vacuum pumps, one diffusion pump capable of attaining pressures less than 10 microns in the vacuum and rocket thrust chamber, and a GN_2 ejection system used to purge the vacuum chamber and thrust chamber of volatile combustion products and unreacted propellant.

The GN_2 pressurization system is made up of everything to the right of the dashed line in Figure 4. It maintains a constant pressure in the propellant tanks during the duration of a test run, and it also provides a means of safely relieving dangerous overpressures in the propellant tanks through a pair of blow-out disks.

The third and most important system contains the thruster and the propellant injection assembly (Figures 4 and 5). To include all of the levels of the parameters specified in the test matrix, it was necessary to design eight thrust chambers, having diameters between 1.25 and 4.125 inches and lengths between 0.42 and 16.00 inches, and from four nozzles having contraction ratios of 3 and 9. Sixteen chambers were actually fabricated, eight of stainless steel and eight of acrylic.

Two injector types were included in the parameter test matrix, a like-on-like, self-impinging doublet and an unlike-on-unlike, single-element doublet. The self-impinging injector was designed to obtain the effects of stream interaction upon spray formation for a singly flowed propellant. Although the self-impinging injector was designed to have the same manifold volume, impingement length, liquid injection velocity, and impingement angle as the single-element doublet injector, the vaporization chamber pressure rise times were markedly different for the two injectors. The data obtained using the self-impinging injectors would be the basis against which computed chamber pressurization histories (without chemical reaction) would be compared. The single-element doublet injectors were used for "hot" firings during which oxidizer and fuel were allowed to react.

The propellant solenoid valves are venturi valves manufactured by the Fox Valve Development Company. These stainless steel valves have a pintle with a teflon poppet which provides a seal in the entrance of the venturi section. The propellant volume in the valve downstream of the seat is quite small. The valves were mounted directly to the back face of the injector to minimize dribble volumes (Figure 5). A low-voltage pulse of 24 vdc produced valve opening times of 3 to 4 milliseconds. A high-voltage pulse with pulse shaping reduced the valve opening times to about 1 m/sec.

The flow-controlling orifices and venturis had diameters of 0.028, 0.033, 0.040, and 0.052 inch, corresponding to the four flow rates listed in Table I. The venturis were designed into the propellant valve as described above, with their downstream side fitting snuggly into the injector face. The orifices were designed as separate units to fit between the propellant valves and the injector face, or to be inserted into the injector as were the venturis. The two designs were used to vary the dribble volume. The feed lines from the propellant tanks to the propellant valve were 3/8 or 1/4 inch stainless steel, depending on the line size being considered in the test matrix. The 2-liter propellant tanks were also made of stainless steel.

The quantities measured before and during a test run are listed in Table II along with the measuring and recording device used. The quartz Kistler pressure transducers were used to measure pressure transients because of their rapid response and stability over a large range of operating conditions. The measurements taken between firings were all recorded on strip charts or read from gauges. The transient measurements were recorded on strip charts, magnetic tape, and oscillographic paper. The valve actuation and timing were controlled electronically to within ±2.0 milliseconds. The oxidizer valve was set to lead the fuel by 4 milliseconds. The valve current traces were recorded directly on oscillographic paper, along with the other transient measurements. Once the fire switch was thrown, the propellant valves and recording instruments were all activated automatically by an SEL 810A computer.

PROCEDURE

The testing of each engine configuration involved "cold" and "hot" runs. During cold runs, the same propellant was fed through both the fuel and oxidizer lines and expanded into the thrust chamber. These runs provided information on the pressurization of the thrust chamber when no chemical reaction was taking place. The hot runs involved the use of a fuel and oxidizer, and the rocket was fired in the conventional manner.

Preparatory to each run, hot or cold, the vacuum chamber and thrust chamber were evacuated to a pressure below 10 microns. Once this pressure was reached, the two propellant tanks were pressurized, using the GN_2 pressurization system, to the appropriate tank pressures determined for the flow rates desired. The fire switch was then thrown, activating the valve sequencing system and the recording devices. Two sets of data were taken for each configuration.

RESULTS

Data was obtained for 13 engine configurations all having 1/4 inch propellant valves and feed lines. The magnetic tapes were interpreted by computer and printed out in engineering units.

Analysis of propellant valve current and pressure traces revealed that the valves were not functioning as had been expected. A definite pressure dependence was noticed in the valve response times. For pressures at the high end of the operating range, around 900 psi, the response times approached 15 milliseconds while at low pressures, near 200 psi, the response times were approximately 5 milliseconds. A second crucial discovery was the fact that valve sequencing was not controlled. The oxidizer lead had been set at +4 ±2.0 milliseconds, but the observed lead times were varying between +14 and -10 milliseconds. Since the ignition delays being analyzed were less than 20 milliseconds in duration, these variations in the valve characteristics and sequencing make the use of the data for such an analysis very questionable.

Finally, the chamber pressure traces indicated a significant nonlinear thermal drift in the Kistler transducer output signals at the end of the steady state and continuing into the tail-off portions of the pressure pulses. The drifting was not noticed during each run until after ignition had occurred and the engine had time to heat up. It increased with temperature and usually produced meaningless data during the tail-off period.

An inspection of the propellant valves disclosed two major defects. The valve poppets had been shortened by constant hammering against the valve seats. Though the teflon poppets were replaced between configurations, the replacement parts were not made with sufficient accuracy for precision operation and the error was perpetuated to succeeding configurations. The internal metal surfaces of the valves showed significant marring, the result of improper alignment of the valve pintle due to the deformed poppet and of constant handling of the valves between configurations. An analysis of the valves by Fox Valve Development Company verified the defects sighted and found the valve responses to be similar to those indicated by the data.

Several interesting observations were made from the data concerning the reactivity of the fuels and the importance o. certain engine parameters. The ignition delays attained using the different fuels indicated the following trend in the ignition delay times: N_2H_4 > MMH ≈ 50/50 > UDMH. This would imply a greater reactivity for the more highly substituted hydrazines. A look at the vapor pressures of these propellants (VP N_2H_4 = 14.19, VP MMH = 49.47, VP UDMH = 167 mm of Hg), however, indicates that the substituted hydrazine vaporizes more readily than hydrazine, causing ignition to occur sooner. The variations in the oxidizer valve lead times which were unintentionally introduced into the data indicated that the pre-ignition delay was shortened and the ignition spiking amplified with increasing lead time. This occurred because the thrust chamber filled rapidly with oxidizer vapors, raising the chamber pressure before the fuel was introduced, so that the pre-ignition reaction time was shortened.

A third conclusion which could be made from the data was that variations in the impingement length did not affect the pre-ignition chamber pressurization. This verifies the conclusion of the Thiokol investigations (References 1 and 2) that this was probably due to the fact that the oxidizer was forming a spray at the injector outlet before impingement occurred.

SECTION IV

CONCLUSIONS AND SUMMARY

CONCLUSIONS

1. An analysis of the data obtained from the Altitude Ignition Program has been made and judged to be too inaccurate to be of value for investigating the ignition delay and other processes occurring in attitude control thrusters. The source of the inaccuracy was pinpointed to the nonrepeatable performance of the propellant valves.

2. Inspection of the valves revealed that the moving parts were worn and did not conform to the manufacturer's specifications. It was also concluded that it would be too costly in time and manpower to resolve the problems and complete the test matrix in time to support the Pulse-Mode Performance Program.

3. The data indicated the following trend in the ignition delay times: N_2H_4 > MMH ≈ 50/50 > UDMH. It also showed that the oxidizer lead time significantly affected the pre-ignition pressure rise, shortening the delay time and amplifying the pressure spiking. The effect of impingement angle and impingement length seemed to be negligible at space ambient conditions.

SUMMARY

The proposed data test matrix to be completed by the in-house effort included the variation of 17 engine parameters using propellant combinations of MMH, N_2H_4, 50/50, and UDMH as fuels and NTO and ClF_5 as oxidizers. This report concludes a 4-month analysis of the resultant data. Inspection of the data revealed that the propellant valve characteristics and the repeatability of the valve opening sequence were both inadequate for obtaining the precision required to analyze the processes which occur during the pre-ignition pressurization of a thruster. Furthermore, the chamber pressure measurements experienced a notable thermal drift near the end of the steady-state portion of a pressure pulse and continuing on into the tail-off portion. In many cases, this drift produced meaningless numbers in the tail-off region. It was concluded that the data could not be used for the purposes for which it was intended.

The facility hardware was analyzed to find the causes of the propellant valve malfunctioning. The inability to obtain repeatable valve action was found to be due to: (1) the deformation of the valve poppets and marring of their surfaces during operation and handling, and (2) a pressure limit on the valves which resulted in a hesitation of the poppets at pressures near the high end of the operating range.

It was concluded that the in-house program be discontinued since the time needed to resolve the problems which were uncovered and to retake the data would be too long to provide any support to the Pulse-Mode Performance Model Program.

TEST MATRIX FOR CHAMBER PRESSURIZATION STUDY

	X_1	X_2	X_3	X_4	X_5	X_6	X_7	X_8	X_9	X_{10}	X_{11}	X_{12}	X_{13}	X_{14}	X_{15}	X_{16}	X_{17}
1	+	+	+	+	+	+	+	+	+	+	+	+	+	+	+	+	+
2	+	+	+	+	-	-	+	-	-	-	-	+	+	+	+	-	-
3	+	+	+	-	+	-	-	+	-	-	-	+	-	-	-	+	-
4	+	+	+	-	-	+	-	-	+	+	+	+	-	-	-	-	+
5	+	+	-	+	+	-	-	-	+	-	-	-	+	-	-	-	-
6	+	+	-	+	-	+	-	+	-	+	+	-	+	-	-	+	+
7	+	+	-	-	+	+	+	-	-	+	+	-	-	+	+	-	+
8	+	+	-	-	-	-	+	+	+	-	-	-	-	+	+	+	-
9	+	-	+	+	+	-	-	-	-	+	-	-	-	+	-	+	+
10	+	-	+	+	-	+	-	+	+	-	+	-	-	+	-	-	-
11	+	-	+	-	+	+	+	-	+	-	+	-	+	-	+	+	-
12	+	-	+	-	-	-	+	+	-	+	-	-	+	-	+	-	+
13	+	-	-	+	+	+	-	+	-	-	+	+	-	-	+	-	-
14	+	-	-	+	-	-	+	-	+	+	-	+	-	-	+	+	+
15	+	-	-	-	+	-	-	+	+	+	-	+	+	+	-	-	+
16	+	-	-	-	-	+	-	-	-	-	+	+	+	+	-	+	-
17	-	+	+	+	+	-	-	-	-	-	+	-	-	-	+	-	+
18	-	+	+	+	-	+	-	+	+	+	-	-	-	-	+	+	-
19	-	+	+	-	+	+	+	-	+	+	-	-	+	+	-	-	-
20	-	+	+	-	-	-	+	+	-	-	+	-	+	+	-	+	+
21	-	+	-	+	+	+	+	+	-	+	-	+	-	+	-	+	-
22	-	+	-	+	-	-	+	-	+	-	+	+	-	+	-	-	+
23	-	+	-	-	+	-	-	+	+	-	+	+	+	-	+	+	+
24	-	+	-	-	-	+	-	-	-	+	-	+	+	-	+	-	-
25	-	-	+	+	+	-	+	+	+	-	-	+	+	-	-	-	+
26	-	-	+	+	-	-	+	-	-	+	+	+	+	-	-	+	-
27	-	-	+	-	+	-	-	+	-	-	+	+	-	+	+	-	-
28	-	-	+	-	-	+	-	-	+	-	-	+	-	+	+	+	+
29	-	-	-	+	+	-	-	-	+	+	+	-	+	+	+	+	-
30	-	-	-	+	-	+	-	+	-	-	-	-	+	+	+	-	+
31	-	-	-	-	+	+	+	-	-	-	-	-	-	-	-	+	+
32	-	-	-	-	-	-	+	+	+	+	+	-	-	-	-	-	-
33	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
34	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
36	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

X_1 Propellants
X_2 Flow Rates
X_3 A_t
X_4 L*
X_5 Contr Ratio
X_6 Temp Cond
X_7 Flow Control
X_8 Valve Dyn
X_9 Feed Sys Size
X_{10} Press Drop
X_{11} Inj Type
X_{12} Dribble Vol
X_{13} Inj S/V Ratio
X_{14} Imp ∠
X_{15} Imp Length
X_{16} Inj Vel
X_{17} Chamb Mt'l

Figure 1. Test Matrix for Chamber Pressurization

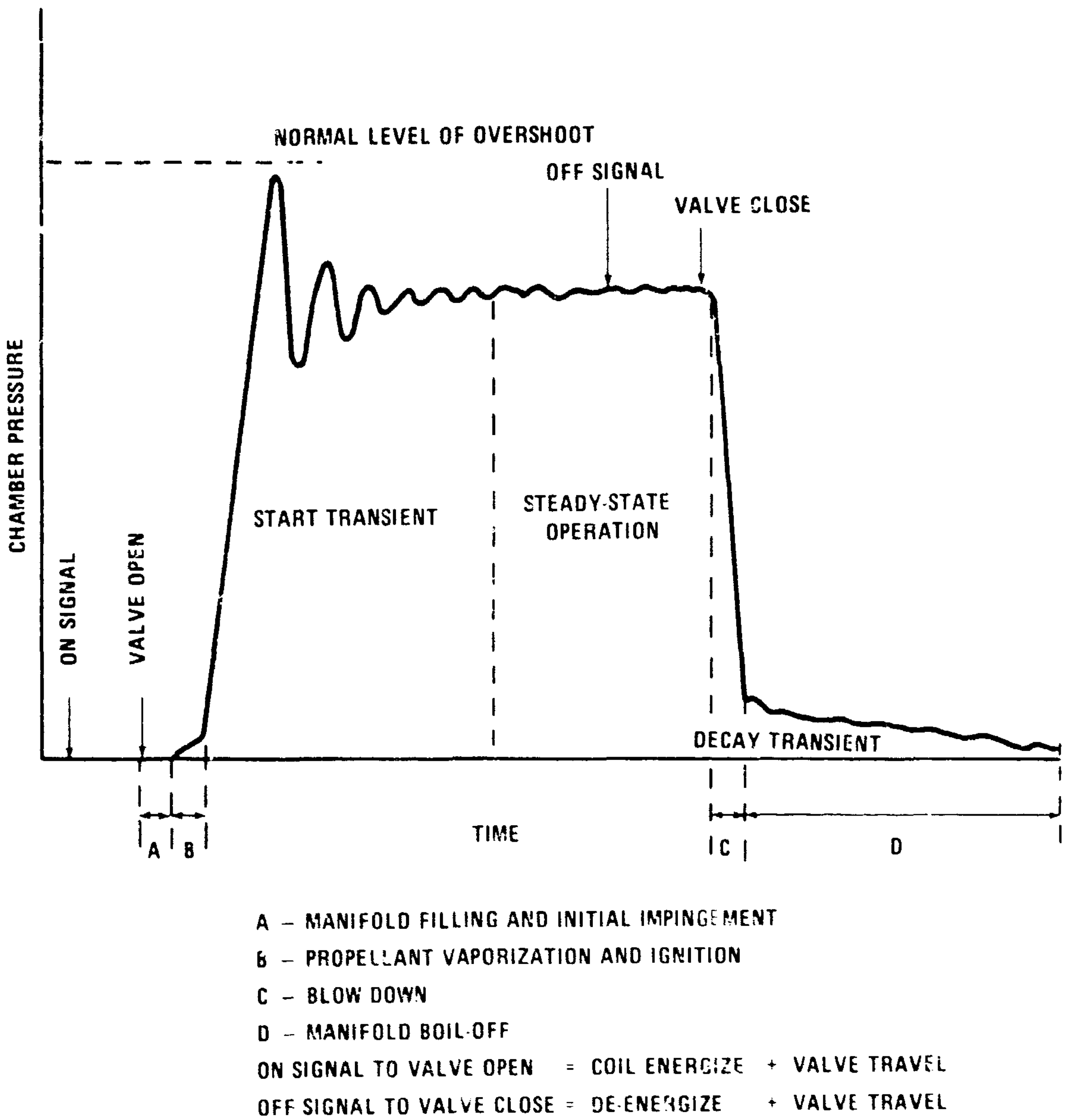

Figure 2. Schematic of a Typical Pulse Trace

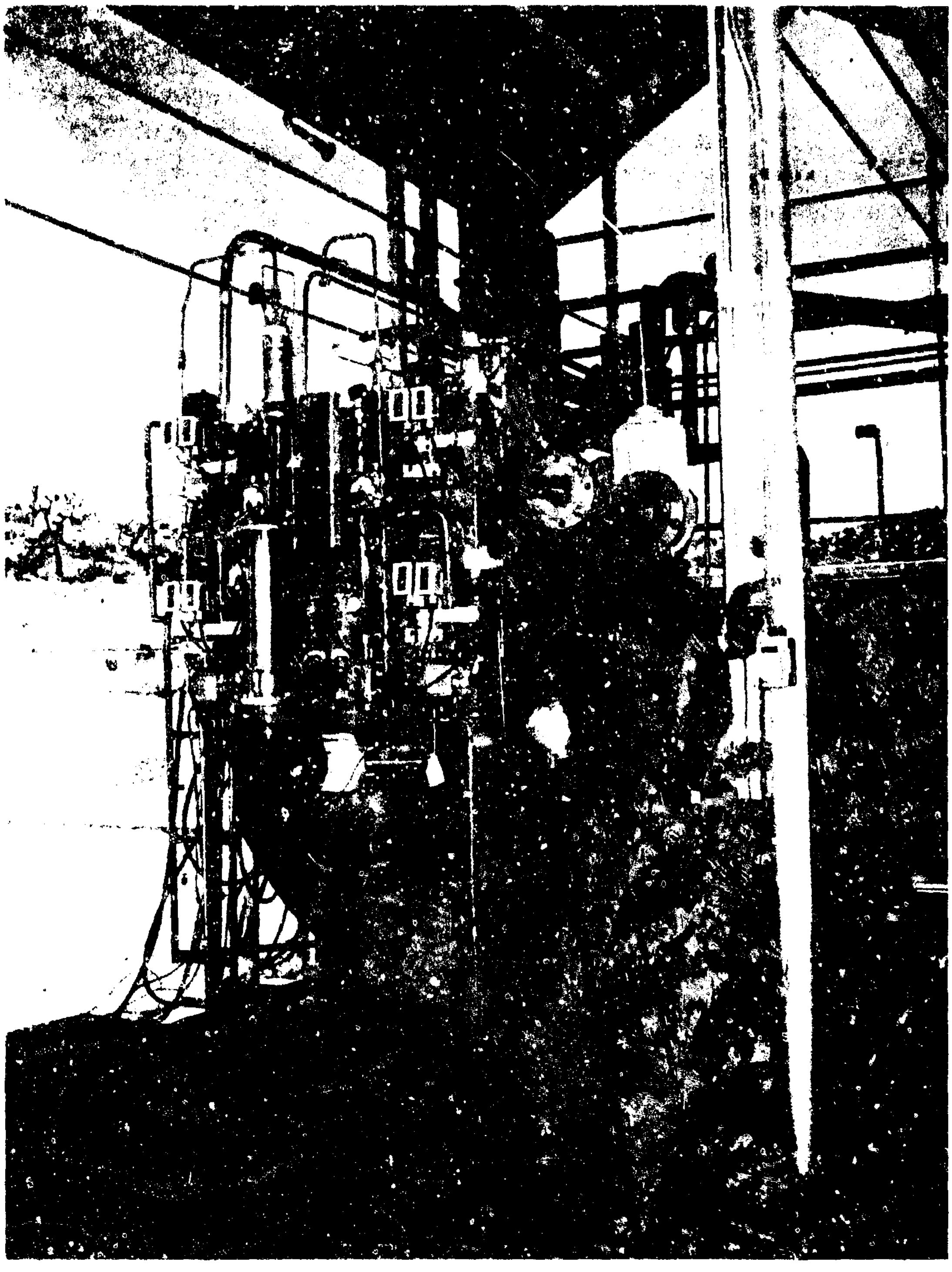

Figure 3. Altitude Ignition Facility

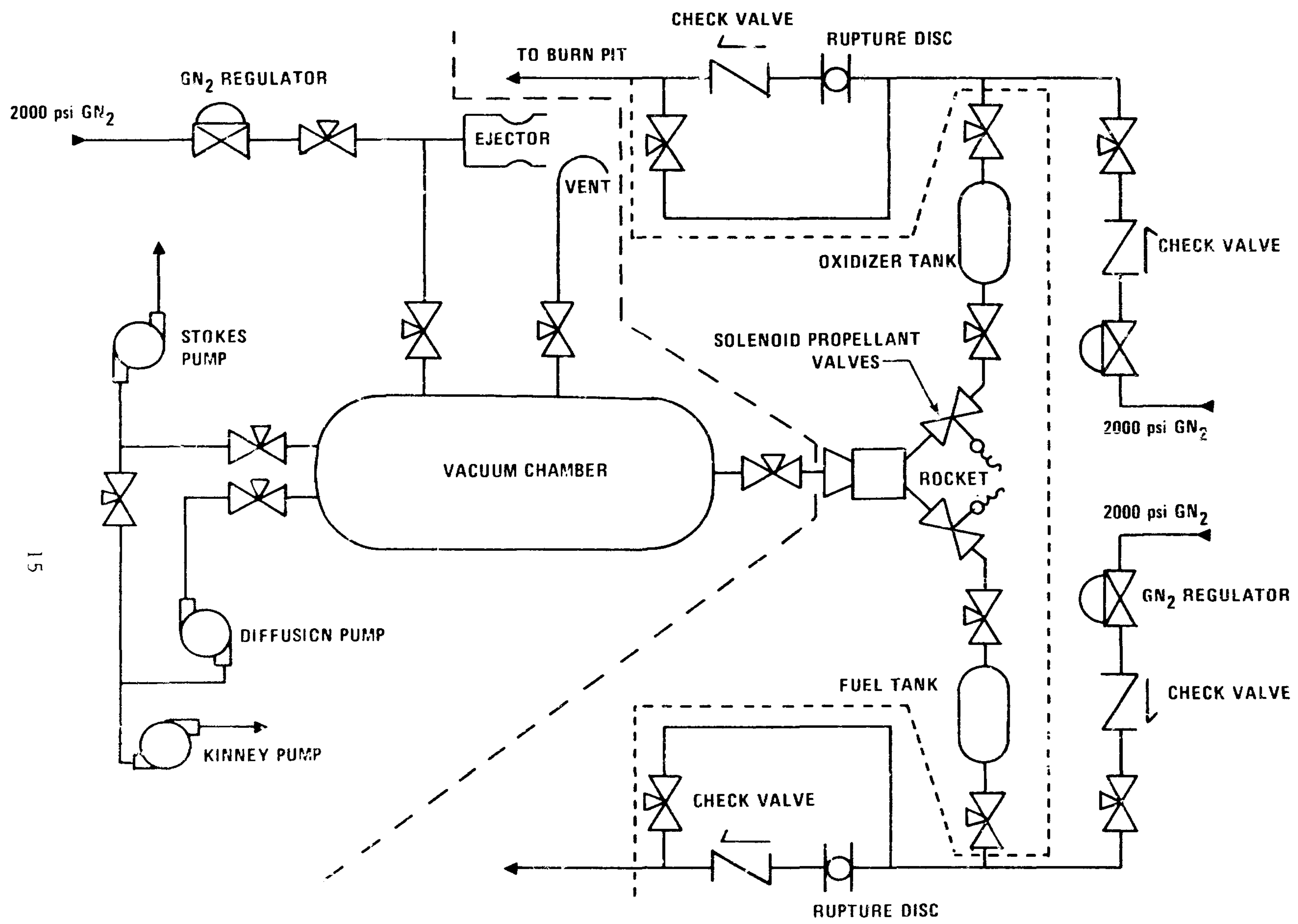

Figure 4. Schematic of the Altitude Ignition Facility

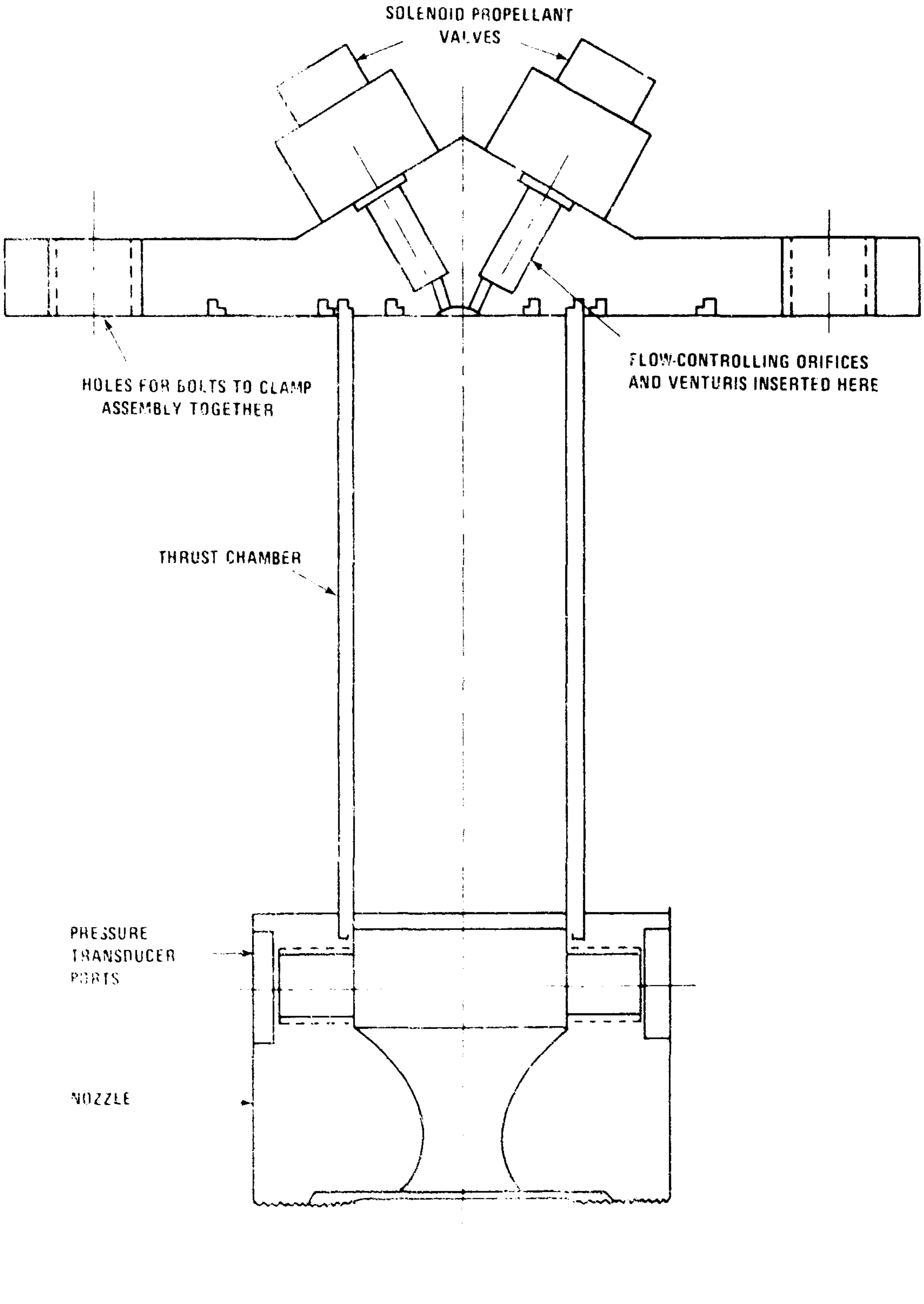

Figure 5. Thruster and Propellant Injection System

TABLE I. ENGINE PARAMETERS INCLUDED IN THE TEST MATRIX

	Parameter	(+) Parameter Levels	(-)
X_1	Propellant	NTO	MMH, UDMH, 50/50, N_2H_4
X_2	Flowrates NTO (lbm/sec) MMH, etc. (lbm/sec)	0.120 0.075	0.240 0.150
X_3	Nozzle Throat Area (in.^{2})	1.253	0.272
X_4	L* (inches)	10	50
X_5	Contraction Ratio	3	9
X_6	Propellant and Hardware Temperature (^oF)	20	90
X_7	Flow Control	Cavitating venturi	Square-edged orifice
X_8	Valve Dynamics	Rated voltage (24 vdc)	Rated voltage (100 vdc)
X_9	Feed System Dynamics (Valve and Line Size), (inches)	1/4	3/8
X_{10}	Pressure Drop Compensation	With accumulator	Without
X_{11}	Injector Type	Unlike-on-unlike (single-element doublet)	Like-on-like (self-impinging doublet)
X_{12}	Dribble Volume (cu in.)	0.01	0.04
X_{13}	Injector Manifold Surface-to-Volume Ratio	To be determined	During injector design
X_{14}	Impingement Angle (degrees)	45	90
X_{15}	Impingement Length (inches)	0.14	0.030
X_{16}	Injection Velocity (ft/sec)	40 80	80 160
X_{17}	Material of Chamber	Acrylic	Stainless steel

TABLE II. ALTITUDE IGNITION FACILITY
INSTRUMENTATION DESCRIPTION

Measured Quantity	Measuring Device	Range	Recording Device	Comments
Vacuum Chamber Pressure	Mechanical gage Cha-Thermocouple gage Cha-Ionization gage	0 to 15 psia 1 to 1000 microns To 10 microns	Gage Gage Gage	Must be less than 10 microns before run
GN_2 Supply Pressure	Mechanical gage	0 to 3000 psig	Gage	Held at 2000 psi
Oxidizer Tank Pressure	Mechanical gage	0 to 900 psig	Strip chart	Set before run
Fuel Tank Pressure	Mechanical gage	0 to 900 psig	Strip chart	Set before run
Oxidizer Valve Pressure	Kistler 603A transducer	0 to 900 psia	Oscillograph and magnetic tape	Monitored during run
Fuel Valve Pressure	Kistler 603A transducer	0 to 900 psia	Oscillograph and magnetic tape	Monitored during run
Rocket Chamber Pressure	Kistler 603A transducer Kistler 701A	0 to 300 psia 0 to 300 psia	Oscillograph and magnetic tape	Monitored during run
Rocket Chamber Temperature	C/A & I/C thermo-couples	-100 to +6000°F	Strip chart	Monitored during run
Oxidizer Tank Temperature	I/C thermocouple	-100 to +100°F	Strip chart	Monitored during run
Fuel Tank Temperature	I/C thermocouple	-100 to +100°F	Strip chart	Monitored during run
Oxidizer Valve Current	Directly into oscillograph	0 to 5 amperes	Oscillograph	Monitored during run
Fuel Valve Current	Directly into oscillograph	0 to 5 amperes	Oscillograph	Monitored during run

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Professional Experience:

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