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THE Marquardt CORPORATION

VAN NUYS, CALIFORNIA

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THE Marquardt CORPORATION VAN NUYS, CALIFORNIA

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FINAL REPORT

VOLUME 1

APOLLO SM-LM RCS ENGINE DEVELOPMENT PROGRAM SUMMARY REPORT

Contract NAS 9-7281

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PREFACE

This report has been prepared in accordance with the provisions of Task I of Contract NAS9-7281. This contract was written to provide for a Product Improvement Program (PIP) on the Marquardt R-4D Reaction Control System (RCS) Engine. The initial use for the R-4D rocket engine was to provide attitude control, docking, ullage and small delta vee maneuvers for the Apollo Service Module (SM) and Lunar Module (LM). Sixteen R-4D engines, in clusters of four, were mounted on each of these vehicles.

This report summarizes the engine development from SM-LM program inception through Qualification and Post-Qualification testing. The report consists of fifteen chapters. Chapter 1 presents an overall summary of the R-4D engine development program. The remaining fourteen chapters discuss in detail the various aspects with the following categories--Thermal Management, Space Ignition Characteristics, Gas Pressurization Effects, Contamination Control, System Dynamic Effects, Structural Design, Material Selection, Propellant Valve Design, Injector Design, Thrust Chamber Design, Test Facilities and Instrumentation, Test Data Analysis, Flight Test Experience, and Reliability.

This report has been divided into four volumes for ease of handling.

CHAPTER 1

MARQUARDT R-4D ENGINE DEVELOPMENT

by C. W. Newhouse

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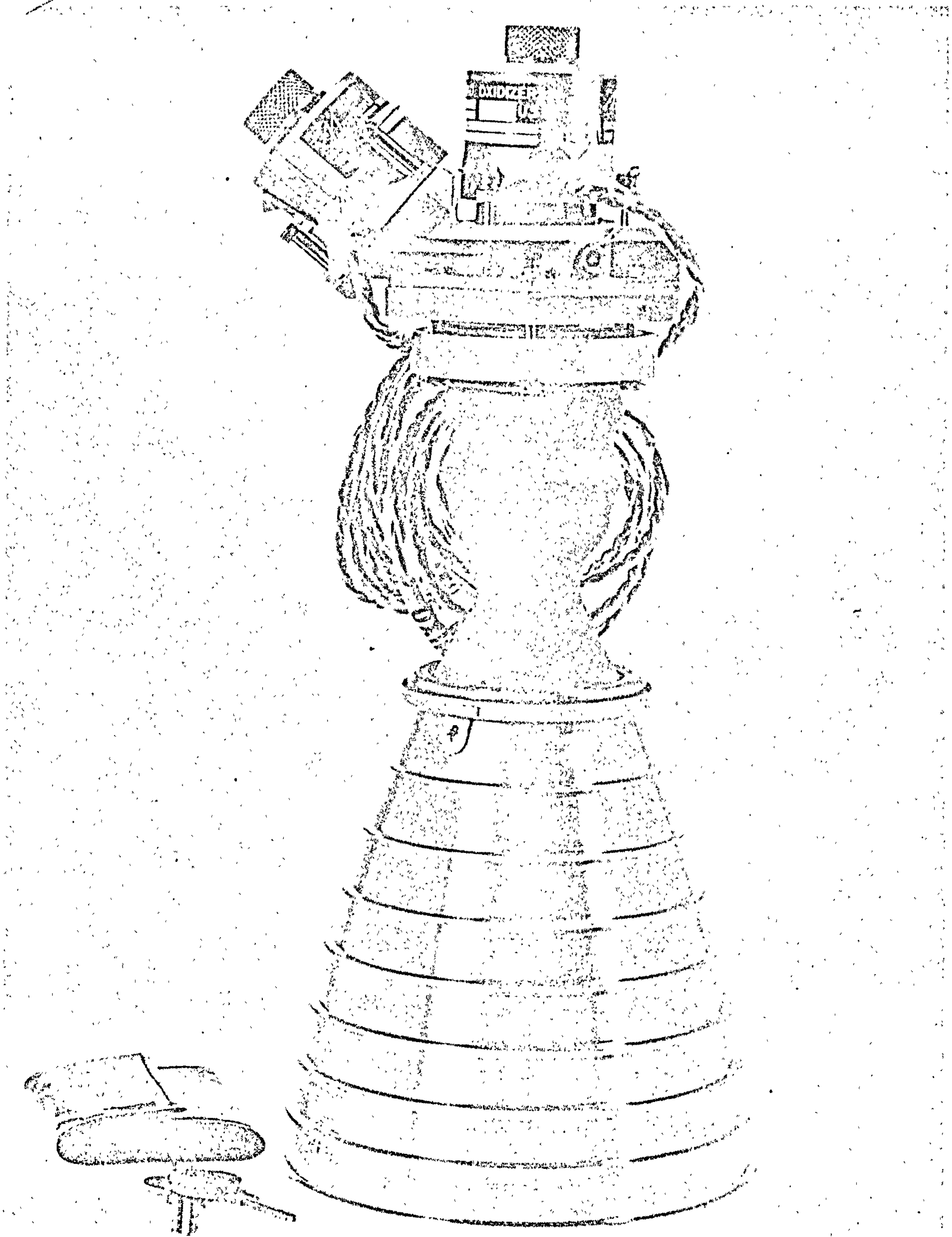
I. INTRODUCTION

Marquardt's R-4D rocket engine has been used on the Apollo Service Module and Lunar Module for attitude control and ΔV maneuvers. The engine was used as the ΔV engine on the Lunar Orbiter and it is presently being qualified for use on other programs. The nominal thrust of the engine is 100-pounds and the propellants are nitrogen tetroxide and monomethylhydrazine or Aerozine-50. The engine is fuel film-radiation cooled and has an extremely long life - projected to be in excess of 100 hours of burn time. Figure 1 is a picture of the engine.

In this chapter, the development of the engine is discussed. Development of the engine started in February 1962. The engine was qualified in December of 1965. The development of the engine can be divided into four main periods: 1) An early development period when different engine configurations were tried and the two major development problems were discovered. These were hot phase burning that burned through the combustors, and ignition overpressures that broke combustors on start-up. A large part of the development effort was spent in solving these two problems; 2) the second period was concerned with the fuel film cooled engine development. By spraying fuel onto the chamber and by building an oxidizer valve standoff into the injector, the hot phase problem was solved. During this period the characteristics of the fuel cooled engine were optimized and the ignition overpressurization problem was being attacked; 3) The third period was devoted to the development of the preigniter that eliminated the ignition overpressurization, and 4) during the fourth period, the engine was qualified and its operating characteristics better defined.



R-4D engine

Figure 1

II. EARLY DEVELOPMENT

Initial go-ahead for the Apollo Program occurred the latter part of February, 1962. North American Aviation, Inc., Space and Information Systems Division (NAA/S&ID) now North American Rockwell Corporation, Space Division placed a TWX order with TMC to proceed with the design, development and production of the Service Module Reaction Control System Rocket engines. This was later superceded by letter contract M2H43X-406013 on 7 May 1962 and by formal contract M4J7XA-406013 on 18 February 1964.

The engine was to be a 100 pound thrust, radiation cooled rocket engine to be used as a pulse modulated, pressure fed, hypergolic bi-propellant engine. It was to consist of an injector head, a thrust chamber and two propellant control valves. The hypergolic propellants were nitrogen tetroxide (N_2O_4) as the oxidizer and a 50-50 blend of unsymmetrical dimethyl hydrazine (UDMH) and hydrazine (N_2H_4) as the fuel (commonly known as A-50).

Initial effort was expended in (1) defining a detailed engine procurement specification with NAA/S&ID, (2) initial analysis phase to provide design criteria for the first phase experimental engine hardware, and (3) initial program planning and preparation of test plans.

The initial program and schedule is presented in Figure 2 and had the basic phases of:

Experimental Tests
Prototype Design Tests
Final Design Tests
Preliminary Flight Rating Tests
Qualification Tests
Formal Reliability Demonstration Tests

A. Experimental & Prototype Stage

During the experimental phase, the analysis, design and development test concepts were directed toward defining the prototype design criteria for the complete engine including the required interface with NAA/S&ID.

The prototype design phase incorporated and formalized the design criteria gained in the Experimental phase into an engine design that was to be developed, tested and proven as the design, which with minor modification would be the configuration for the final design and subsequent Flight Rating Testing and Qualification. Qualification was to be followed by Formal Reliability Demonstration Testing (later deleted from the program).

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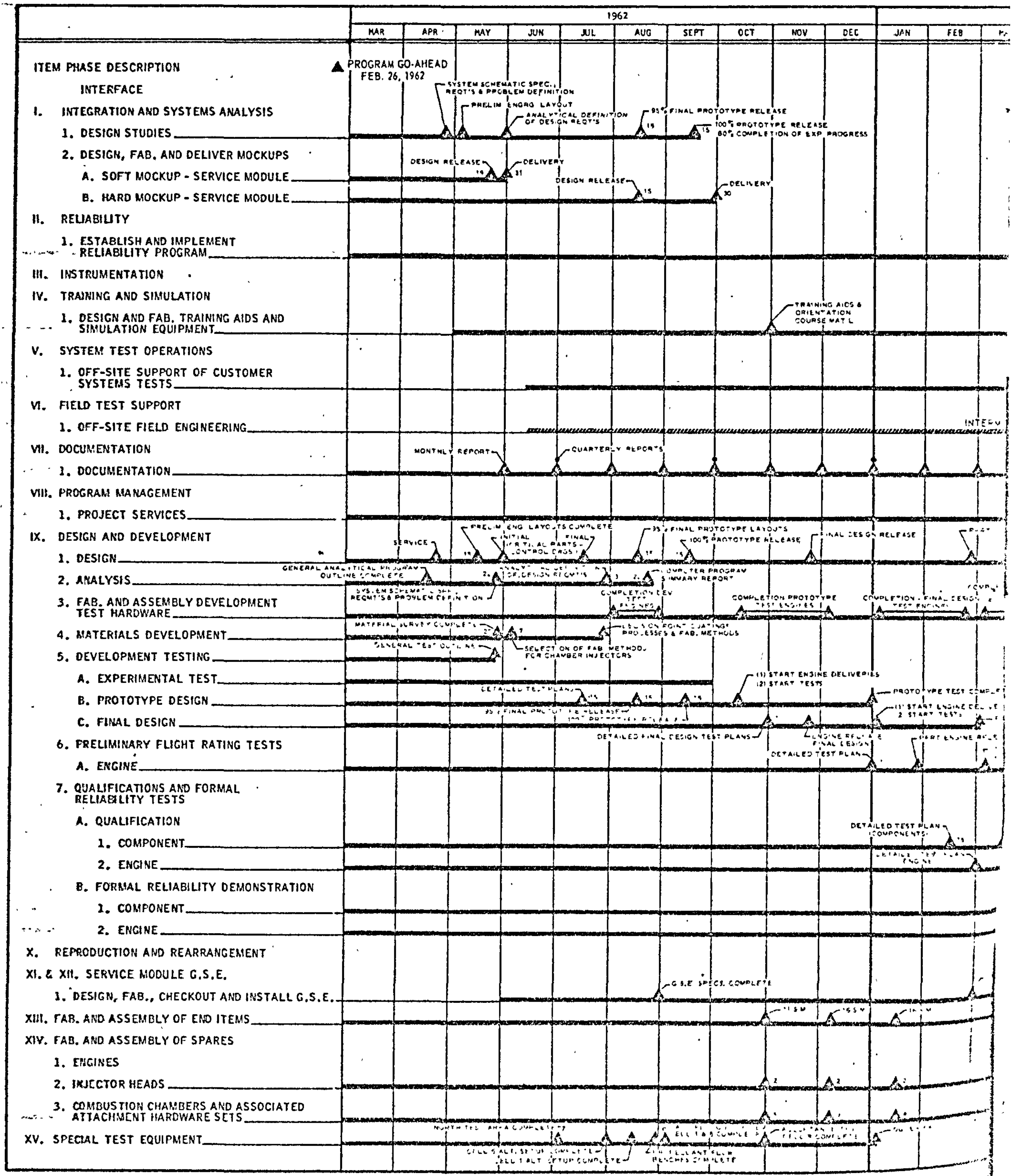
SERVICE MODULE REACTION



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NTROL ROCKET ENGINE PROGRAM

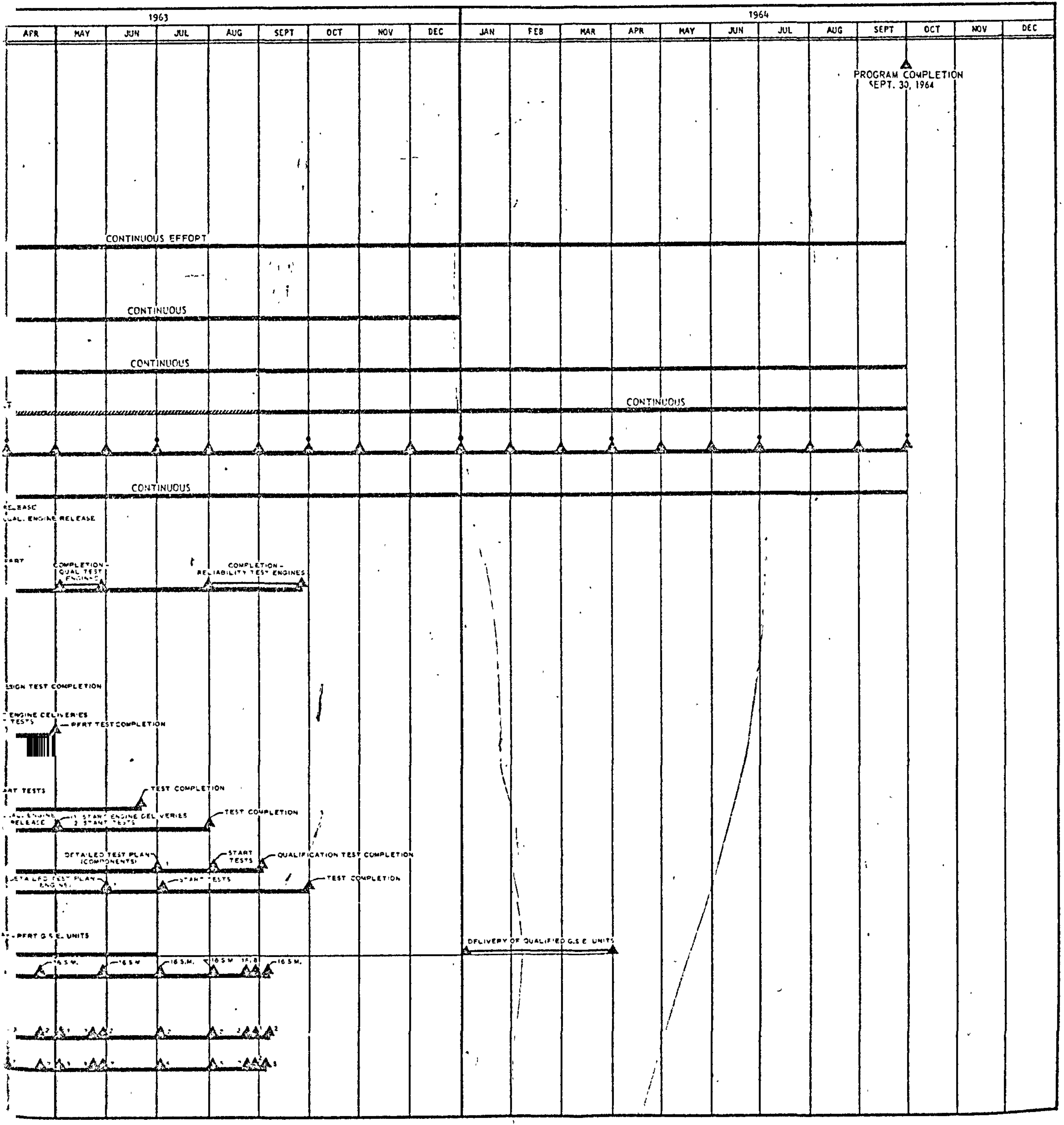


Figure 2

During the latter part of April and early May, 1962, design criteria had been established for the initial experimental hardware and development tests in support of the analysis were being conducted. Utilization was made of existing 25 pound and 100 pound thrust engines from TMC IR&D and earlier development programs.

The first 100 pound thrust sea level engine firing test was made on 17 May 1962 in TMC's MJL Cell No. 1.

Figure 3 presents a schematic of the SMRCS engine during this initial phase.

Initial injector concepts evolved around single doublets in an effort to maintain low dribble volume injectors and to minimize dribble volume effects on engine short pulse performance.

During the summer and fall of 1962, design criteria was being finalized toward the prototype configuration. Many injector configurations were tested and an eight-on-eight multiple doublet injector with oxidizer being injected from the inner holes and fuel from the outer holes was selected for the prototype configuration.

The Experimental Phase was completed basically on schedule and the Prototype Design was released. This engine is shown in Figure 4.

This prototype engine demonstrated good performance I_{sp}'s of 295 and 297. However, its maximum performance, I_{sp} slightly over 310 seconds was not obtained at the desired O/F setting of 2.0 but rather at approximately an O/F of 1.9. Continued parallel analysis, design fabrication and test effort was expended toward increasing peak performance at an O/F of 2.0 while environmental tests were being conducted on the prototype engines. In addition, attempts were being made toward lowering operating and soakback temperatures.

The initial solenoid valve design had three coils, a pull-in coil, a holding coil and an emergency coil.

The initial combustion chamber configuration was a one piece design of coated refractory material, molybdenum disilicide coated molybdenum, with an expansion ratio equal to 40.

Development effort continued on all components of the engine. Flow-turning process for the fabrication of the combustion chamber and exit bell and coating processes were being finalized. The solenoid prototype design was finalized and is shown in Figure 5. Electrical heating and seat leakage tests were conducted substantiating the selection of the configuration. Endurance testing of one million cycles was completed with zero leakage.

INITIAL SM RCS ENGINE SCHEMATIC

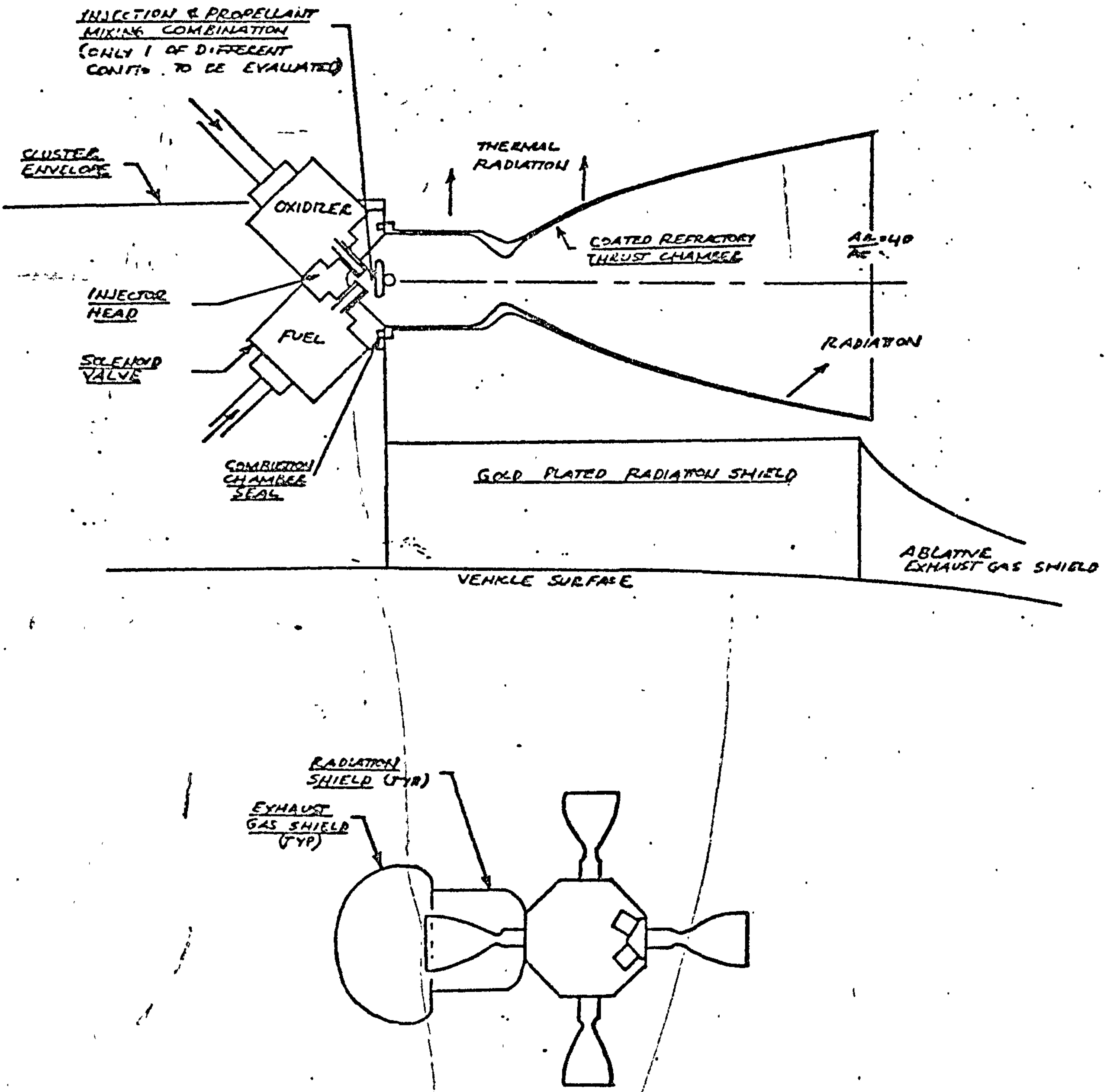


Figure 3

PROTOTYPE DESIGN 225331 .

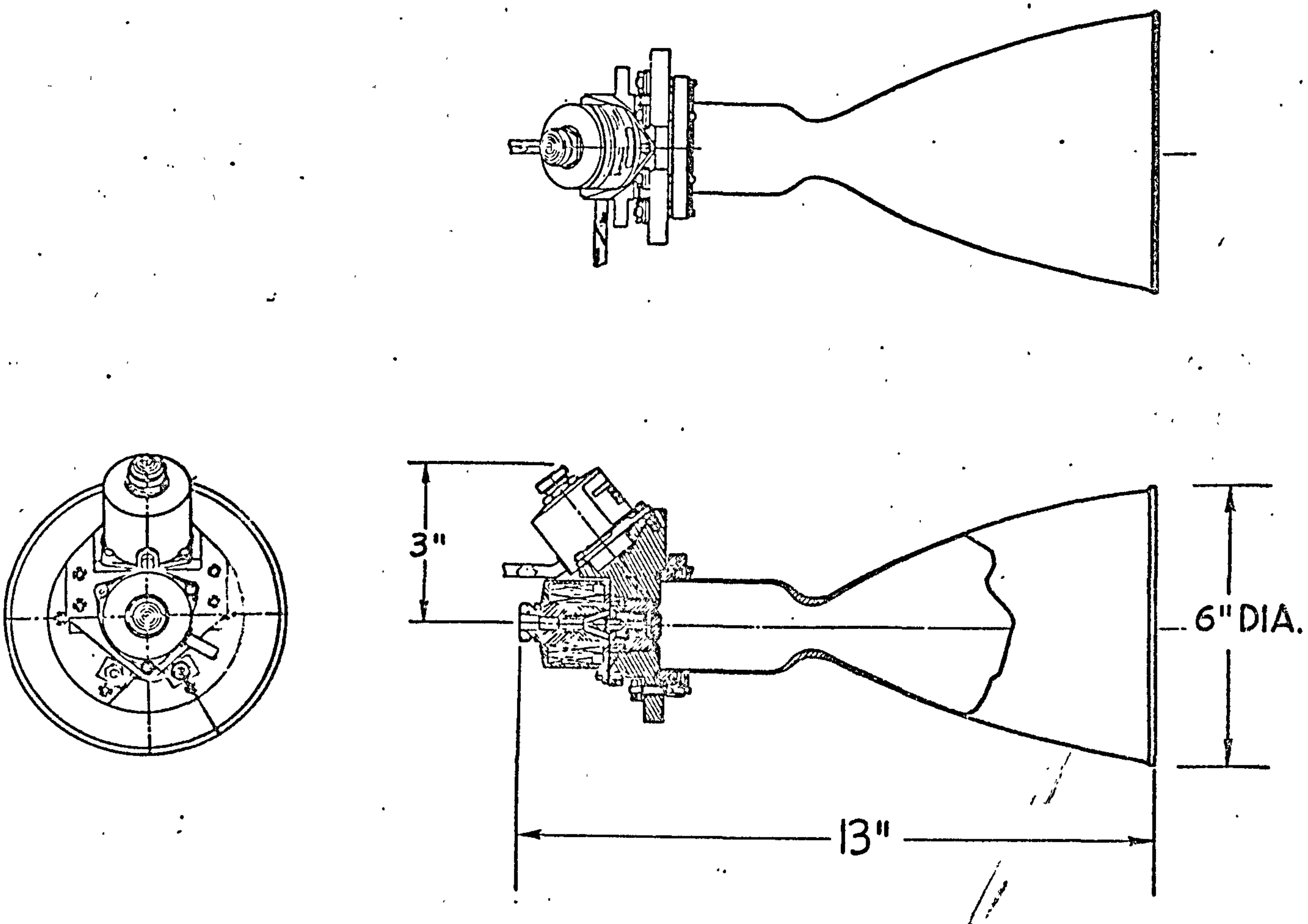


Figure 4

PROTOTYPE SOLENOID VALVE
SERVICE MODULE ENGINE

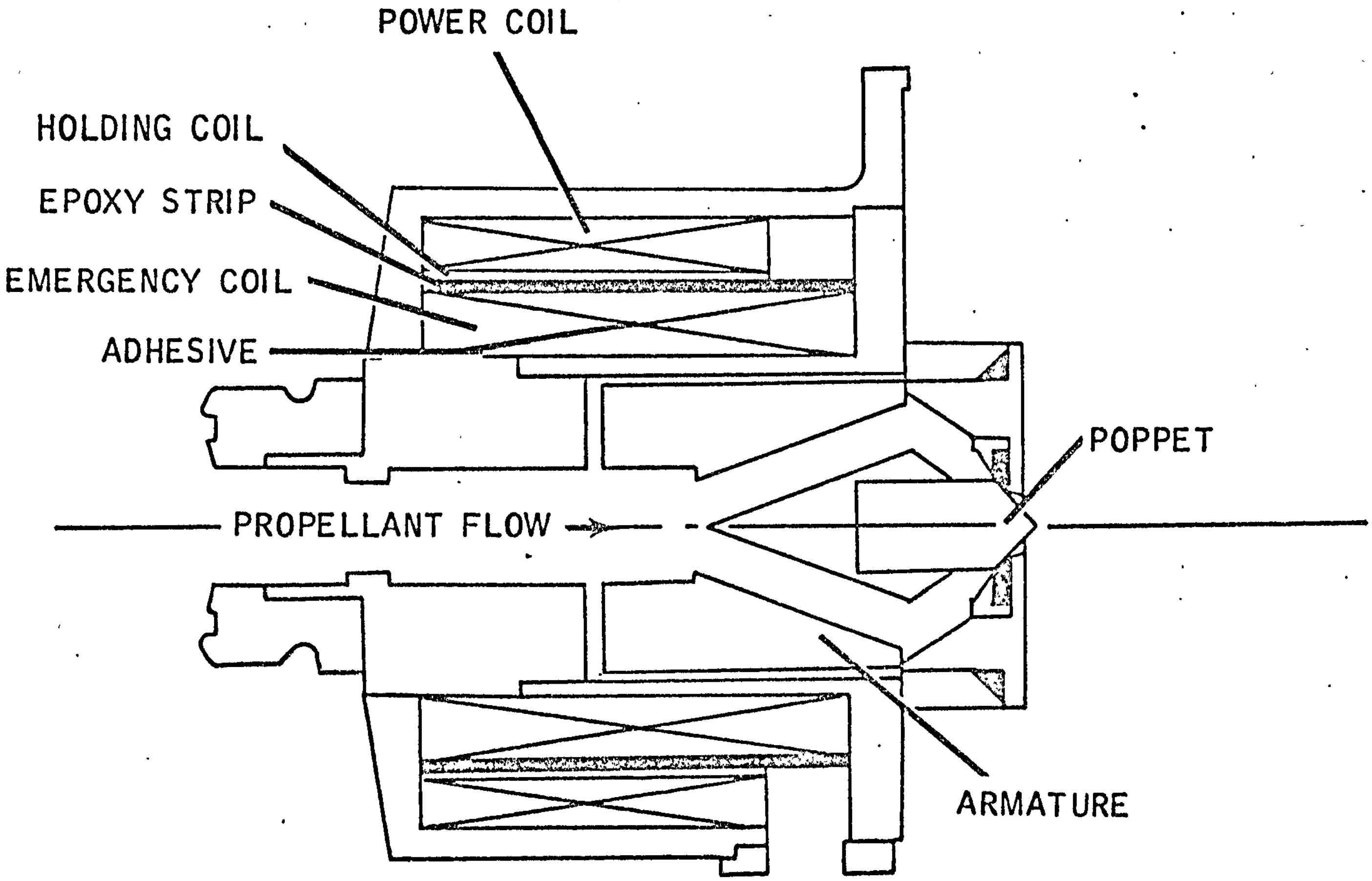


Figure 5

During September 1962, direction was received from NAA/S&ID to change the fuel from A-50 to monomethylhydrazine (MMH). The fuel was changed back to A-50 in November 1962 after damage to the plant which produced MMH. Comparison tests were run between the two fuels and, as can be seen on Figure 6, the difference between them was very small.

In November 1962, NAA/S&ID authorized TMC to design, develop and test a solenoid operated propellant valve with only two coils, one for normal operation and the second for emergency operation. The purpose of this valve was to accomplish a minimum impulse bit with the engine of 0.6 lb.sec. or less. Previously the minimum impulse had been 1.0 lb.sec. TMC proceeded with the requirement and Figure 7 shows the resultant two coil design. One coil, the automatic, is the normal operating coil and the second, manual, is the emergency coil. Extensive sea level, minimum impulse testing was accomplished with the design demonstrating acceptable minimum impulse bit of 0.4 ± 0.2 lb.sec. when converted from sea level to space operation.

Delivery of the first four (4) prototype sea level engines were made during the first week of December 1962, and the first prototype altitude engine was delivered the first week in January, 1963. Figure 8 is a picture of this engine.

Engine testing was being accomplished during this period and Figure 9 shows an altitude engine after it had accumulated 4,381 seconds of burn time during life tests.

Prototype testing was initiated early in 1963 and involved launch vibration and humidity testing. These tests are shown in Figures 10 and 11.

Parallel development tests were continued to better the performance at the design O/F of 2.0 and to reduce operating temperatures. At an O/F of 1.8 to 1.9, injector configurations were demonstrating 295 to 300 seconds of impulse as shown in Figure 12. Operating soakback temperatures were running above design goals.

The flowturn process was selected as the method for fabrication of the combustion chambers and TMC purchased the Flow Turn Machine shown in Figure 13. This machine shear spins the chamber from a preform forging. Once shear formed, the chamber is then heat treated and machined to final dimensions. Previous to this, the chambers were machined from full forgings.

The checkout of Cell 6 was essentially completed in mid-March 1963 and comparison tests with ATL Pad G were conducted. Altitude pressures of less than 0.1 psia were achieved during steady state tests in Cell 6.

THRUST EFFICIENCY vs MIXTURE RATIO

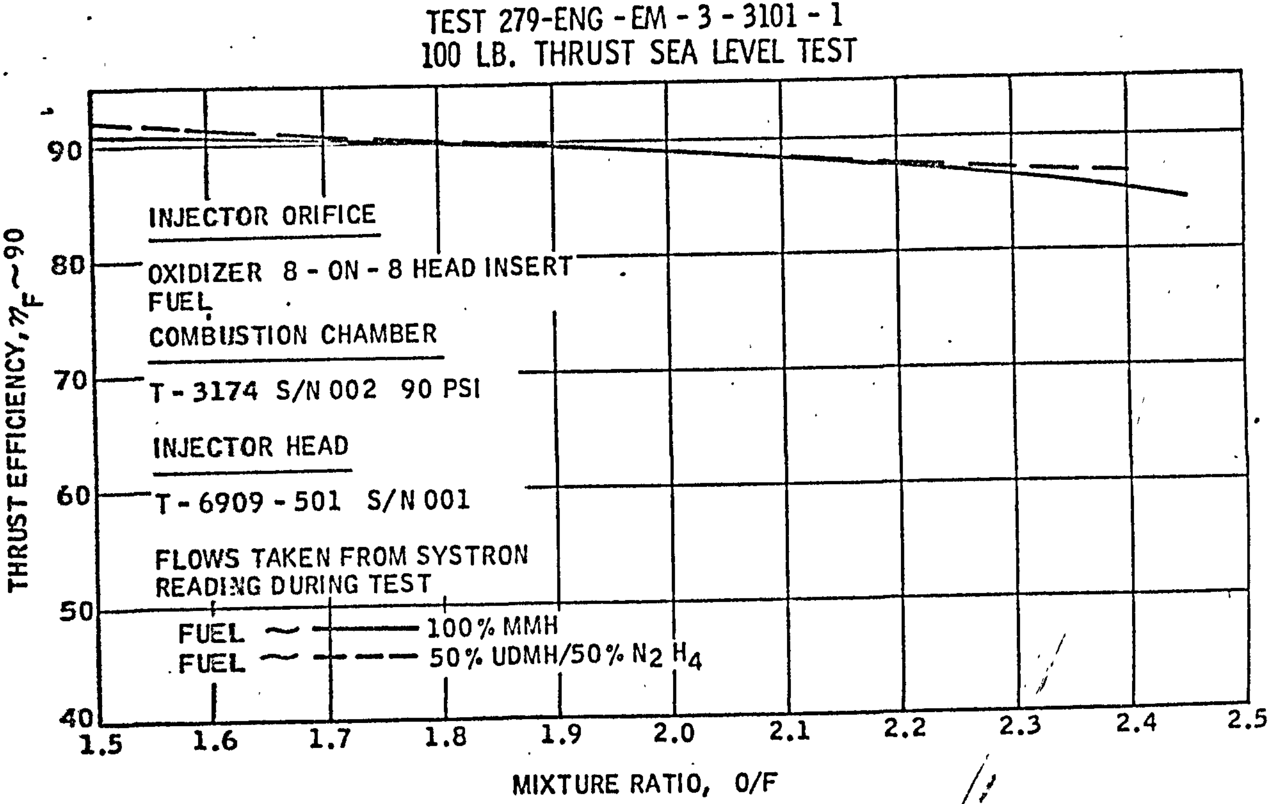


Figure 6

TWO COIL SOLENOID VALVE COIL CONFIGURATION

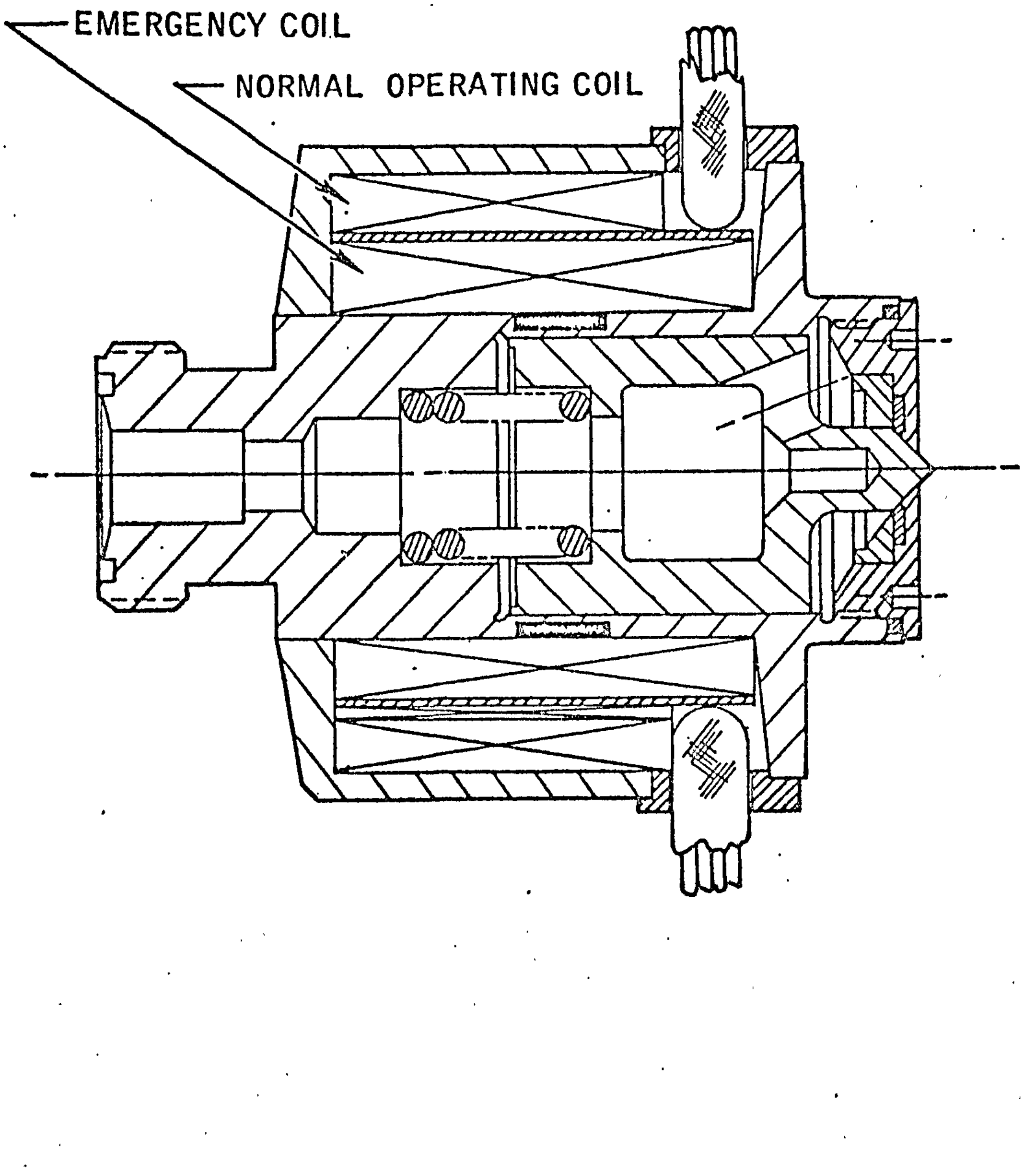
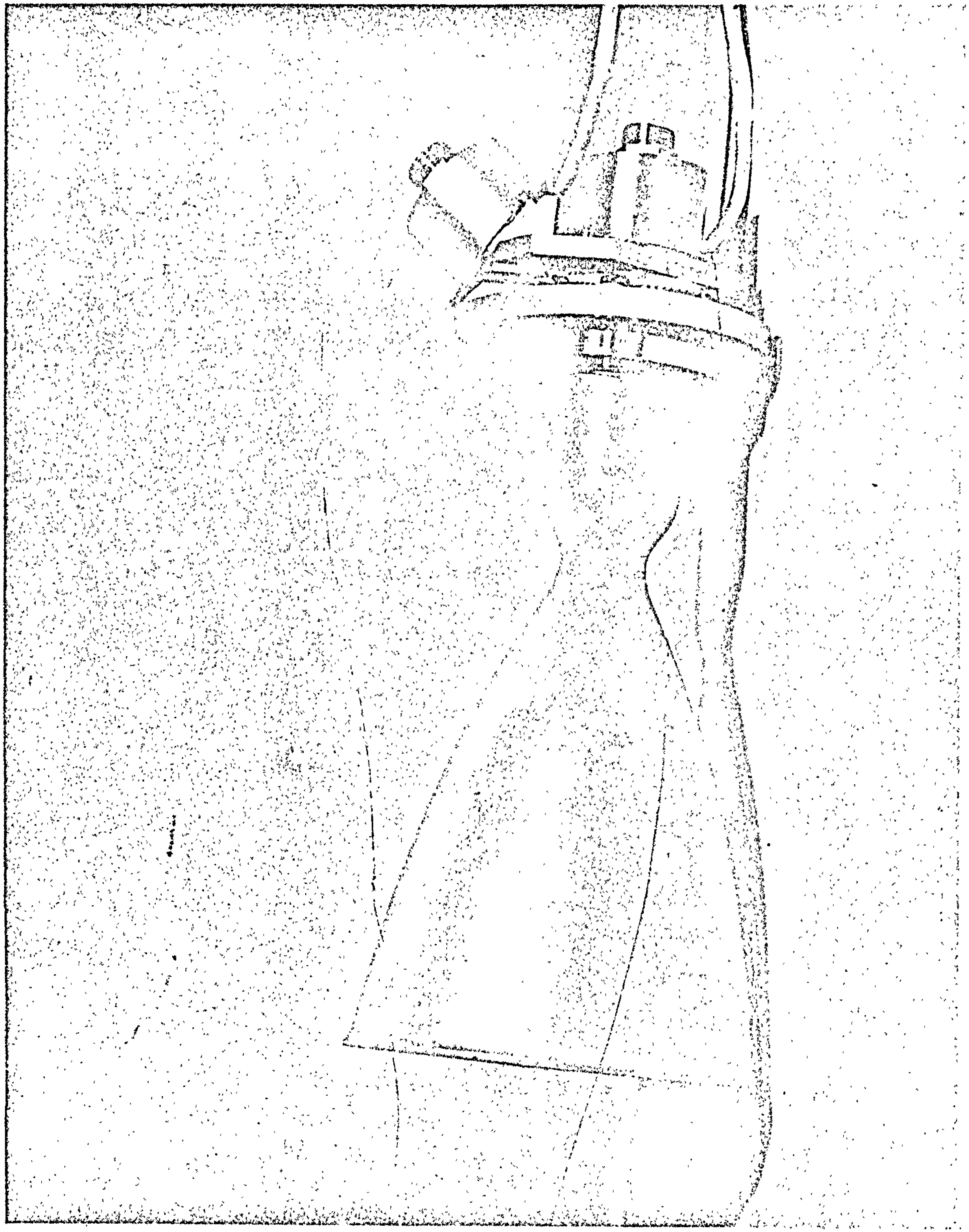


Figure 7



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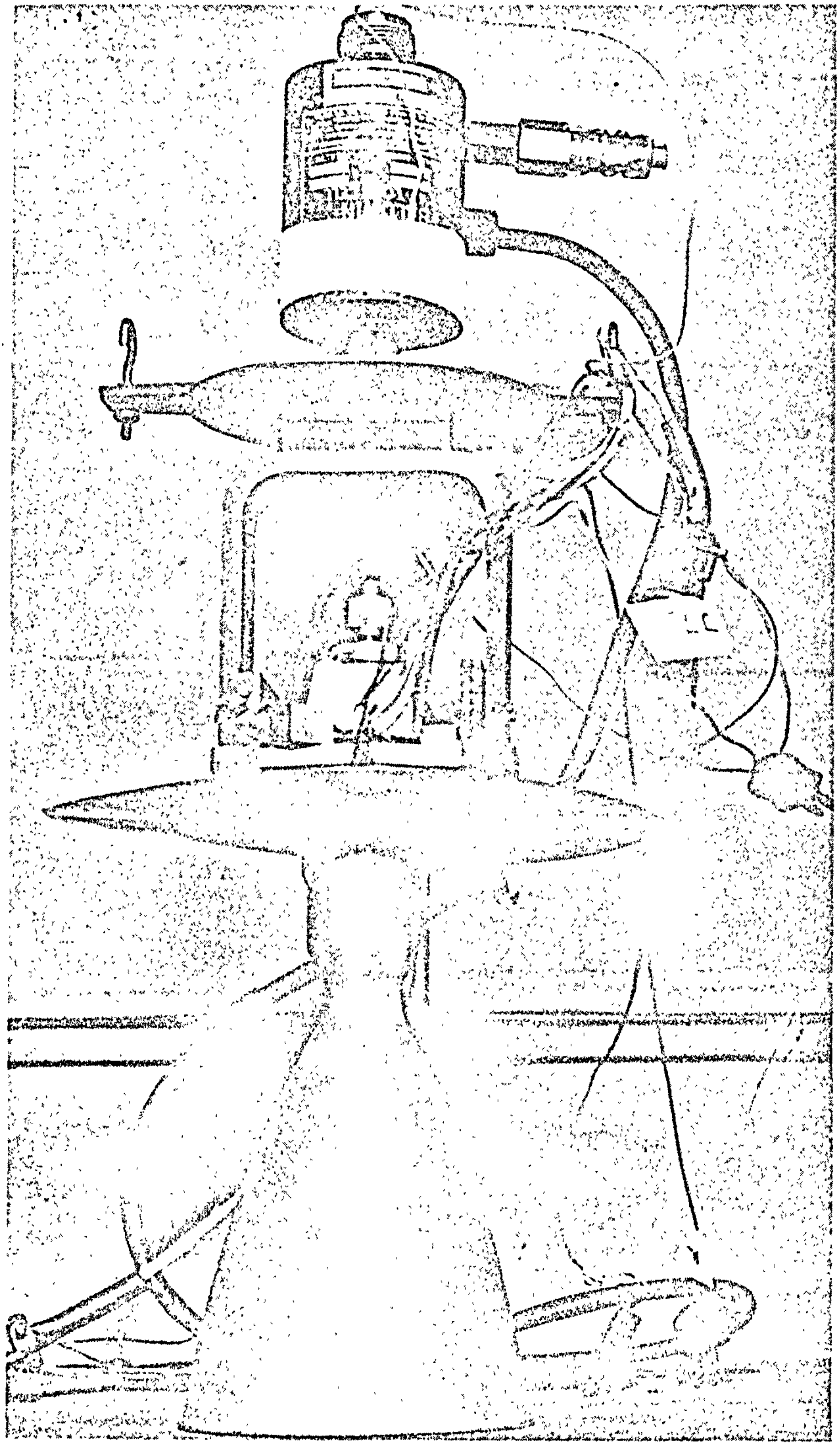
First Prototype Altitude Engine

Figure 8



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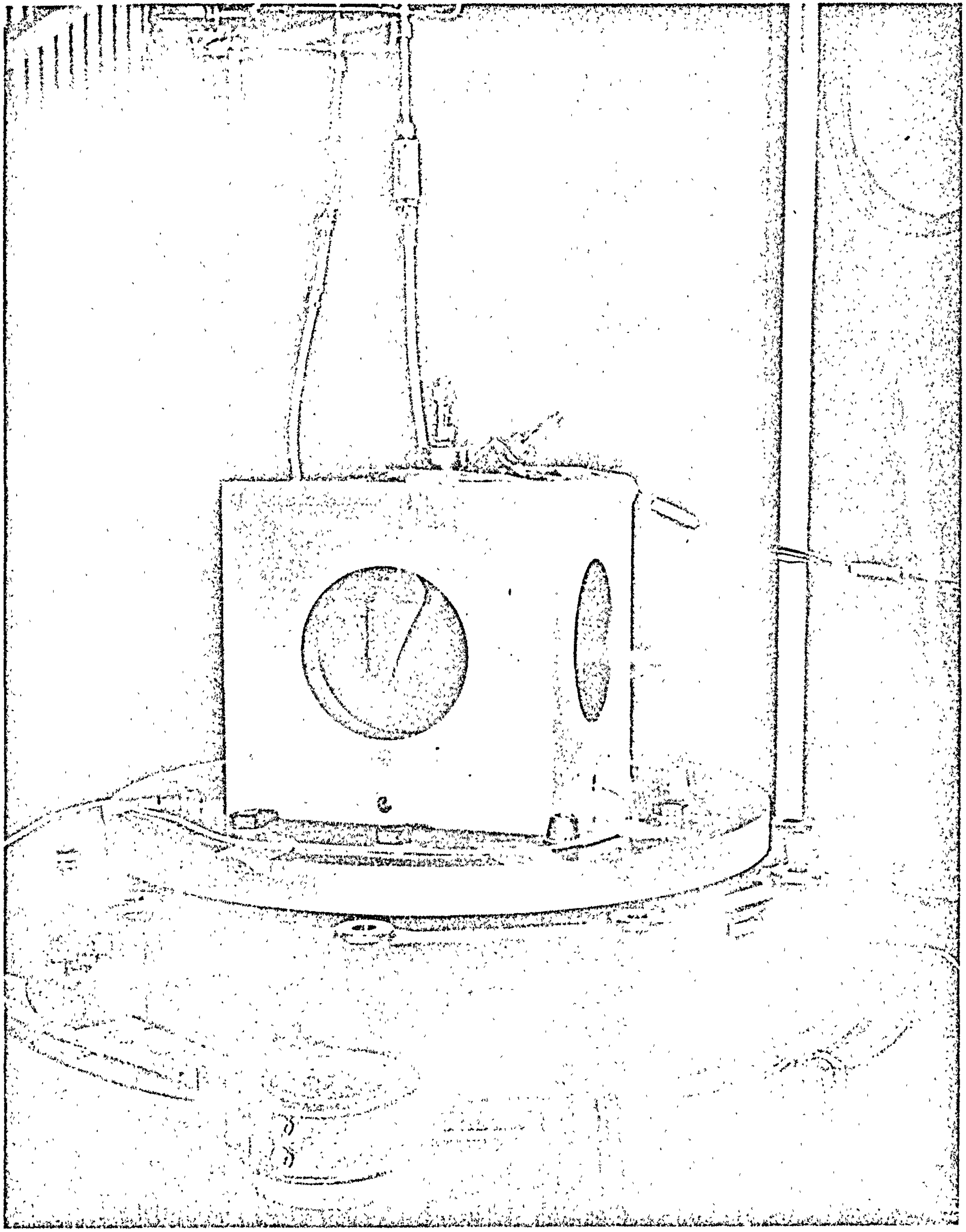
Prototype Engine After 4,381 Seconds of Burn Time

Figure 9



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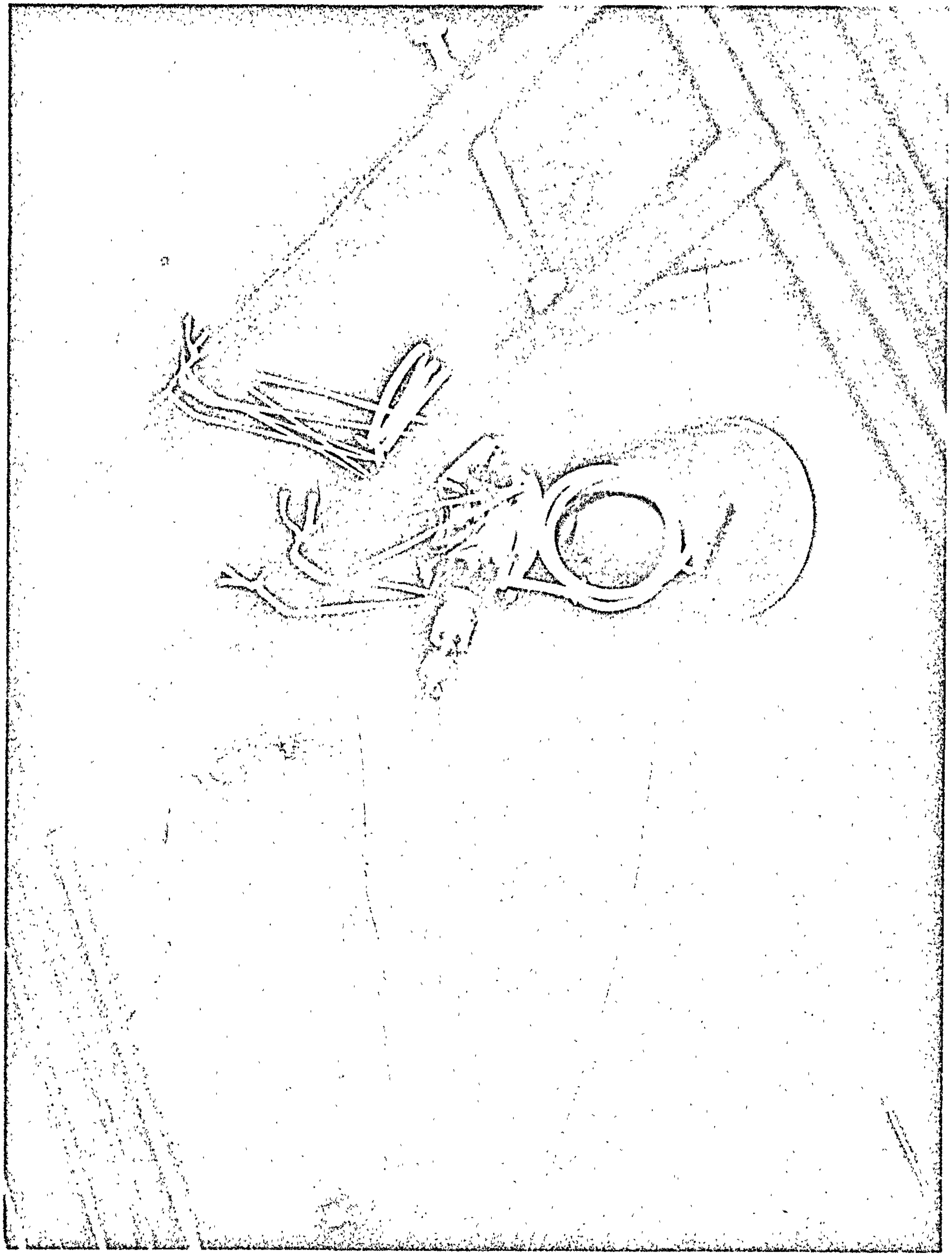
Vibration Test Setup - Prototype

Figure 10



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S/N 002 Prototype Development Engine Suspended in Humidity Box

Figure 11

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SPECIFIC IMPULSE VS MIXTURE RATIO

PROTOTYPE ENGINE

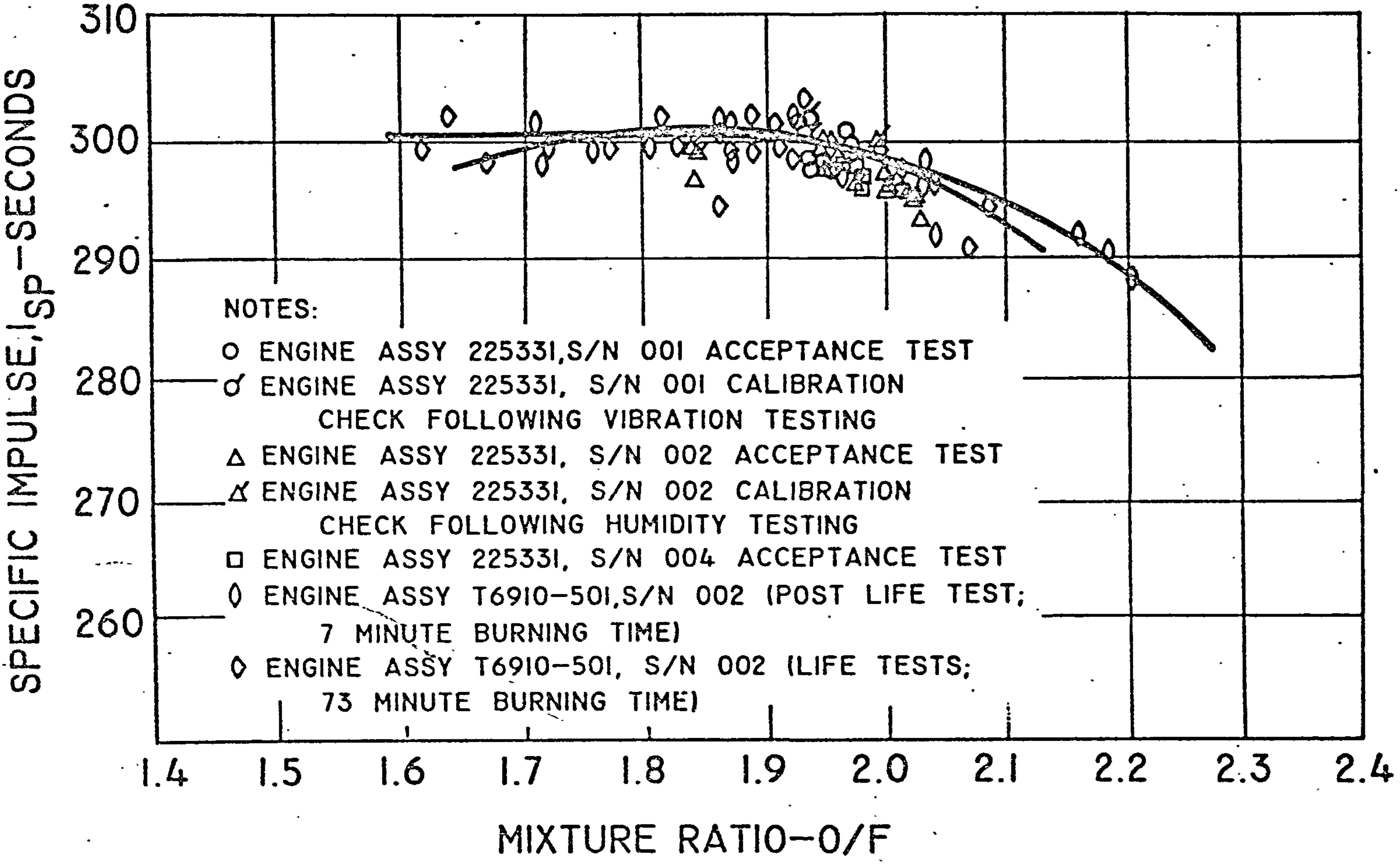
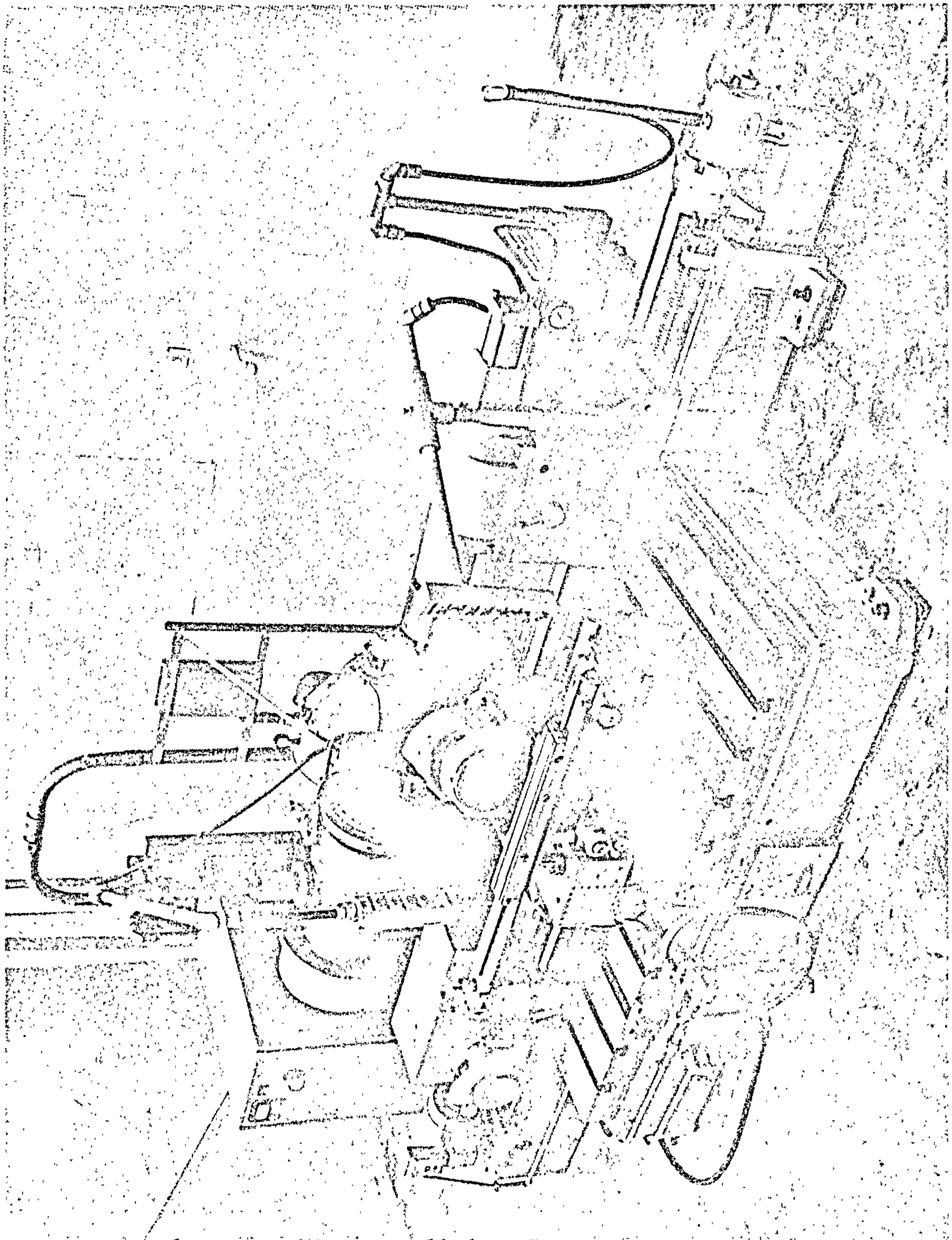


Figure 12



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Floturn Machine Installed at TMC

Figure 13

Testing was continued toward optimizing an injector configuration that would demonstrate Isp performance of 300 seconds or greater at an O/F of 2.0 and an engine configuration that would run cool enough. Various injector to combustor seal configurations were tested in an attempt to increase the thermal resistance between the combustor and the injector head thus lowering the operating and soakback temperatures. In most cases, those seals which did increase the thermal resistance did not provide good sealing of combustor gases.

B. Hot Phase and Ignition Overpressures

The prototype configuration was changed to two coil valves and fuel cooled injector as shown in Figure 14. This engine has steady state Isp performance of 297 seconds. In parallel, a 12 on 12 injector configuration was being tested which demonstrated good performance of 298 to 300 I_{sp} and an O/F of 2.0. Soakback temperatures were approximately 100°F lower than on the previous prototype design.

Tests simulating relief pressure were conducted on the engine. A total of 218 seconds of burn time was accumulated at inlet pressures of 230 psia. Test run times ranged from 3 to 30 second on-times. The engine endured more time than was anticipated at these conditions. The test was terminated when the chamber burned through at the throat in the approximate location of a thermocouple due to the extremely high temperature experienced. in this area (3100°F and possibly higher).

A fuel cooled injector head engine with an 8 on 8 injector was tested in ATL, Pad G to determine soakback temperatures. Soakback temperatures took twice as long to reach oxidizer vapor temperatures as with the previous non fuel cooled heads. A NAA/S&ID mass (simulating the Service Module mounting structure) was then put on the engine and tested. Soakback temperatures were extremely good taking those times as long to reach oxidizer vapor temperature as the first fuel cooled head run.

The engine was then installed in Cell 6 for pulse tests. On June 7, 1963, during these tests, the molybdenum combustion chamber shattered during the conduct of the fourth 5 second steady state run. Excessive oxidizer valve seat leakage was noted during post tests. The engine was refurbished with a new combustor and retested in an attempt to duplicate the previous failure. Twenty three-second steady state runs were conducted successfully. Ten pulses of 10 milliseconds on and 200 milliseconds off were then programmed. During the third pulse of this program, the combustion chamber disintegrated. A post run examination of the injector head and solenoid valves revealed no malfunctions and showed that everything was functioning properly. An investigation was initiated to determine the cause of the explosion. One of the first areas to be explored was that of dribble volume. It was felt that following a normal engine

FUEL COOLED INJECTOR HEAD

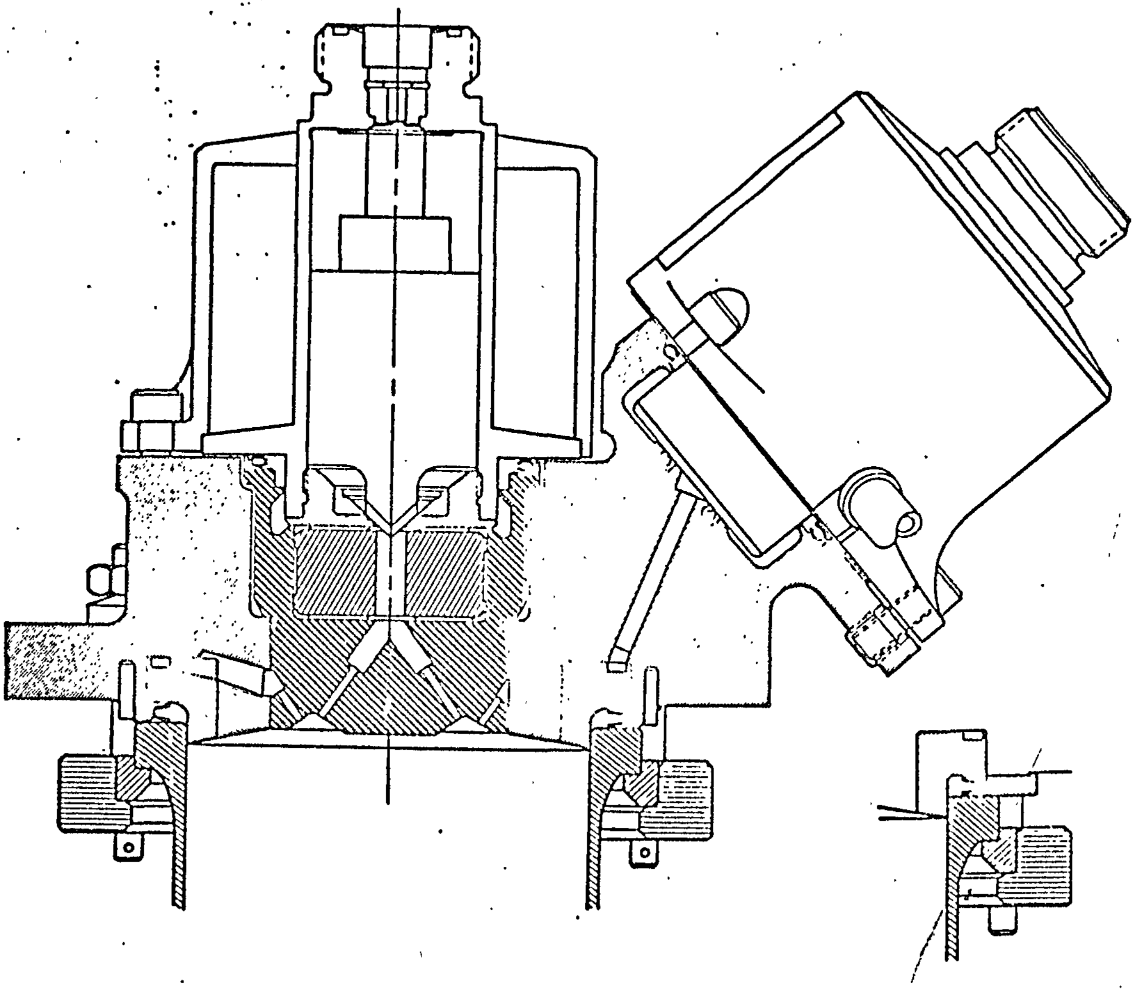


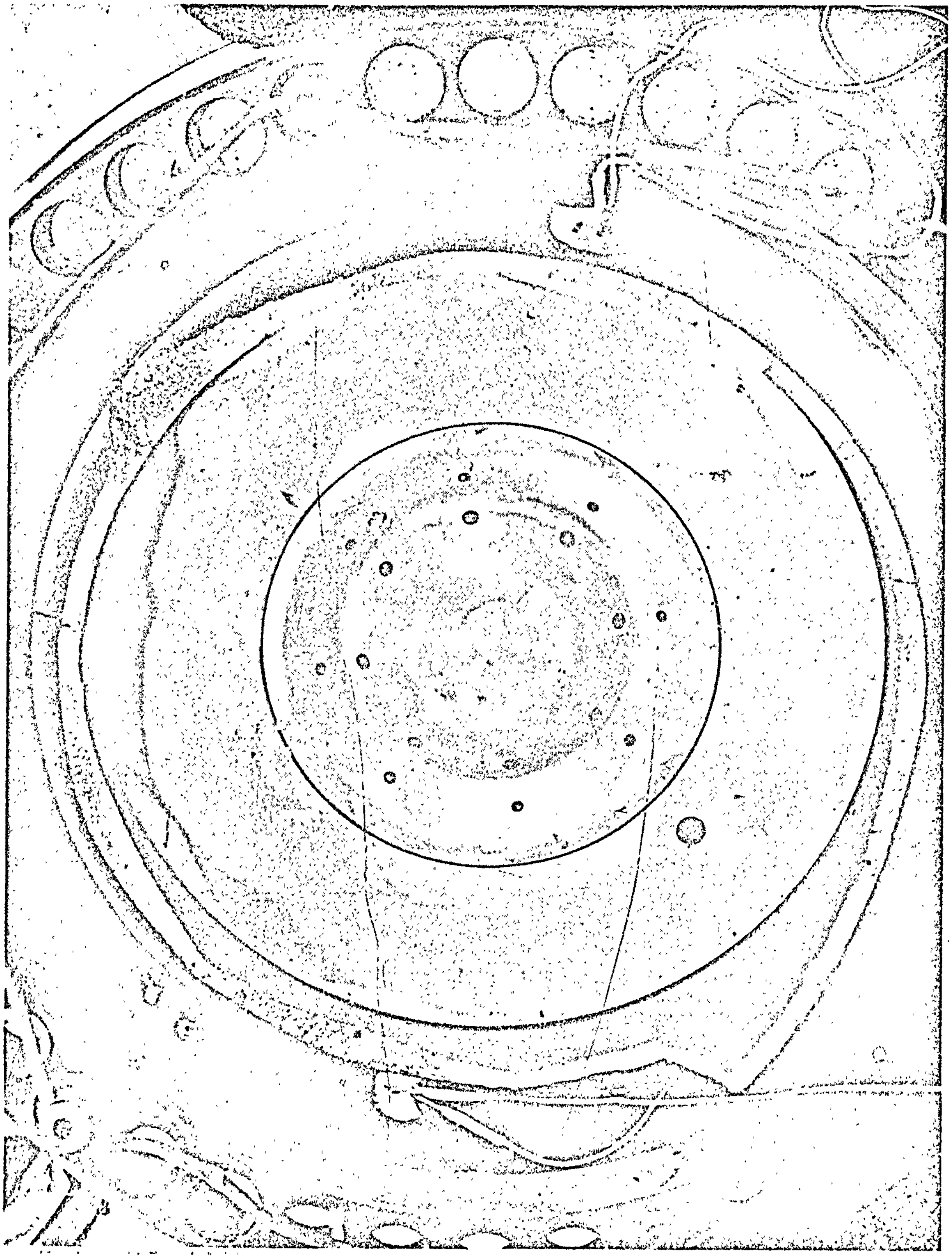
Figure 14

shutdown, the large fuel dribble volume (140% in the fuel cooled head) would slowly exhaust into the combustion chamber and eventually into the facility itself. This fuel would be in a vapor state due to the low cell pressures. Hydrazine compounds in the vapor state have the attributes of a monopropellant. Therefore, it was theorized that when the engine was pulsed again, the hot combustion products would ignite this vaporous hydrazine resulting in a detonation in the combustion chamber.

To investigate this possibility, the engine was again rebuilt to the same fuel cooled head configuration except that a stainless steel combustion chamber with a flush mounted pressure pickup for measuring chamber pressure was substituted for the molybdenum combustion chamber. The tests consisted of six 3 second steady state runs and fourteen pulse series (5 pulses each, 10 to 100 ms on, 20 to 200 ms off) for a total run time of 20 seconds with 76 starts. Although the combustion chamber pressure pickup did not have response rates sufficient to pick up the pressure traces to their absolute peak value, the data definitely did show that a severe pressure spike was present at start-up with this large fuel dribble volume injector head configuration. Although the pressure appeared to peak out at around 300 psia on some of these traces, subsequent investigations indicated that the hydrazine vapor ignited explosively and the pressure transducer used was incapable of recording the actual pressures, which probably were at least an order of magnitude higher than recorded. An investigation of recovered portions of the chambers demonstrated that the chambers had failed due to an evenly applied overpressure condition. These data demonstrated a positive correlation between the severe start-up spikes and the length of time between pulses. Figures 15 and 16 show the broken combustors.

Another series of pulse runs was conducted with this engine to explore failure causation. For these runs, a clear plastic combustion chamber was installed and a high speed camera used to photograph the pulses. The pulses were 250 milliseconds on with varying off times between pulses. The film showed that vapor did collect in the chamber between pulses and did ignite explosively. The program to conclusively demonstrate these failures was actively pursued. However, there was now more than a reasonable certainty that the dribble volume associated with the fuel cooled head was probably the main contributary cause.

It was therefore decided mutually between NAA/S&ID and TMC to halt all development work on the fuel cooled head and investigate another approach that had shown promise in previous tests. A new 12 on 12 injector pattern engine was built-up which included 2 coil valves, dribble volumes of 65% fuel and 122% oxidizer, a stainless steel combustion chamber with a flush mounted P_c transducer, a wide flange "K" seal and no head to chamber bearing. While the starting transient was still present, it did not appear to be of the magnitude of those with the fuel cooled head.



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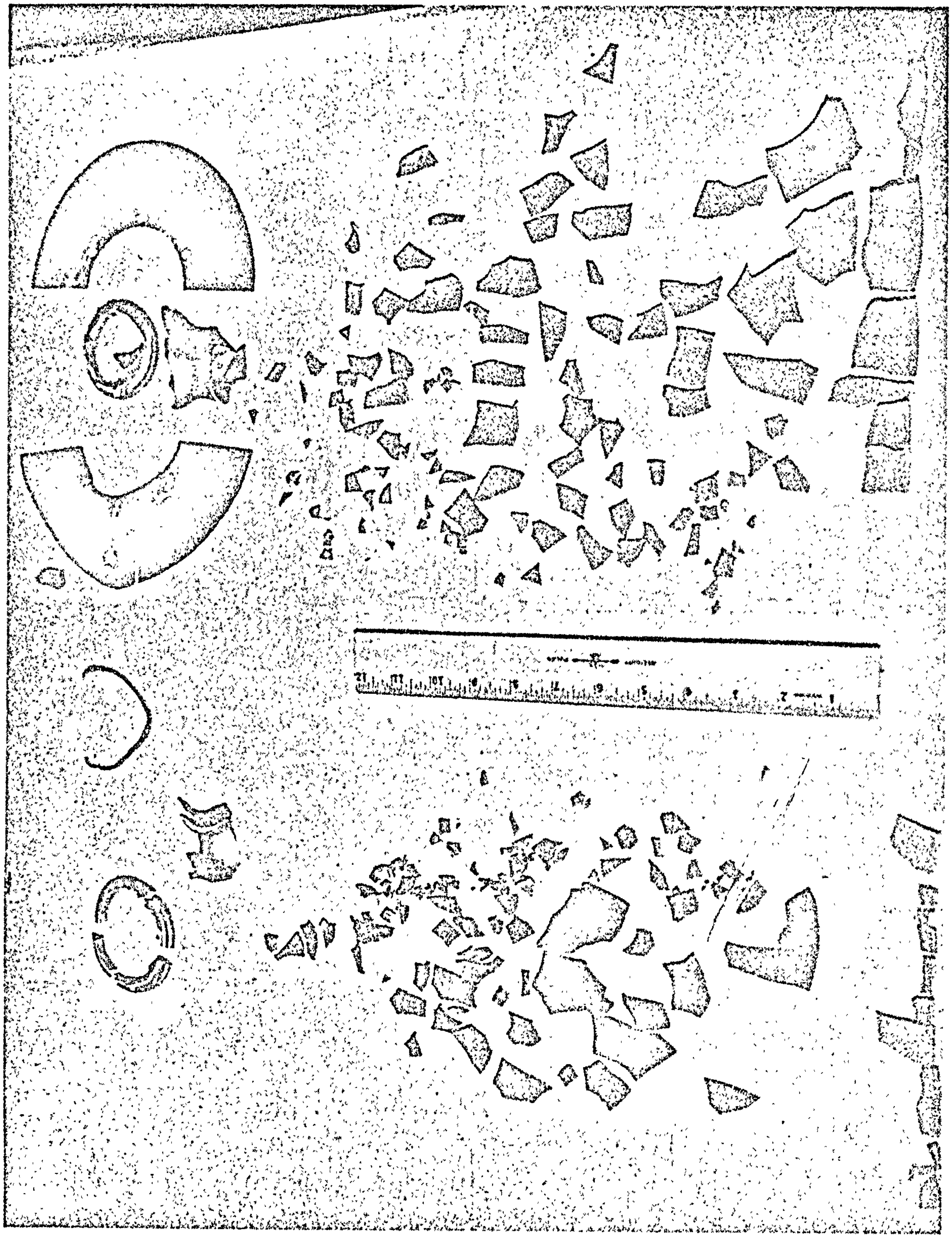
Failure Analysis of Engine

Figure 15



NEG. C9656-2

Fragments of Molybdenum Thrust Chambers

It should be noted that no attempt had been made to optimize the 12 on 12 injector for minimum soakback temperatures during the tests reported immediately above. However, preliminary analyses of the data indicate that this 12 on 12 injector appeared to have soakback characteristics commensurate with those of the fuel cooled head. This injector head design was optimized for soakback tests. The Prototype Development program was restarted on 8 July 1963. The configuration of this engine was a 12 on 12 injector with 71% dribble volume on the oxidizer and 73% dribble volume on the fuel side. Head to chamber bearing was an aluminum ring and the engine had prototype 2 coil valves.

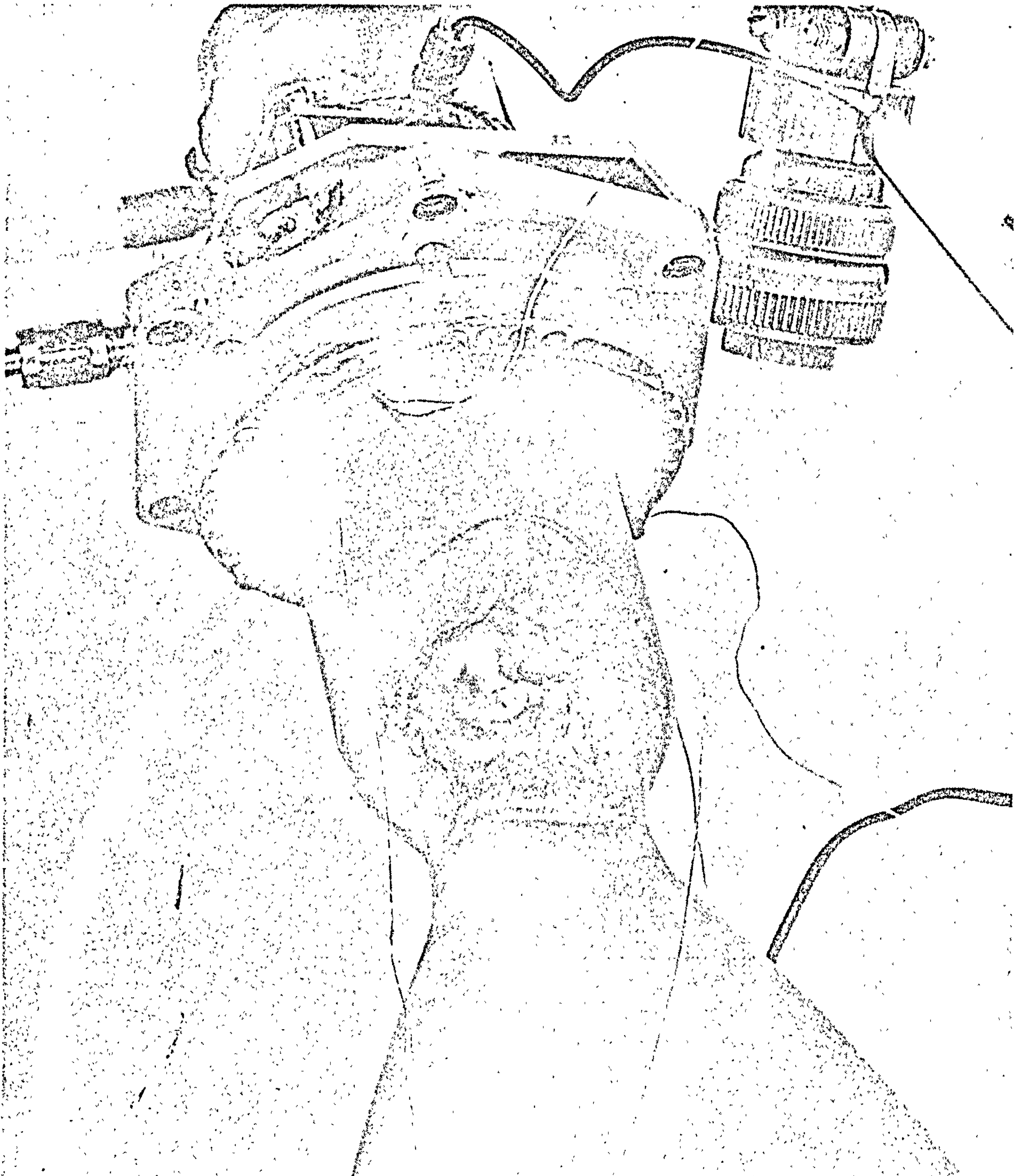
The Prototype Test Plan, Marquardt Test Plan (MTP 0001) consisted of the following appendices:

Appendix A - Steady State and Pulse Tests at Design Conditions
Appendix B - Steady State and Pulse Tests at "Off Design" Conditions
Appendix C - Calibration Tests
Appendix D - Mission Duty Cycle
Appendix E - Low Temperature Tests and Thermal Shock Tests
Appendix F - Life Tests
Appendix G - Malfunction Tests
Appendix I - Vibration, Transport Phase
Appendix J - Vibration, Launch Phase
Appendix K - Vibration, Space Phase in Low Temperature Environment
Appendix L - Shock Tests, Transport Phase
Appendix N - High Temperature Vacuum Tests
Appendix O - Humidity Tests
Appendix Q - Functional Tests

Engine S/N 0004 experienced a combustion chamber burn through apparently due to localized high temperatures in the lower combustion chamber and throat area in excess of the melting point of the molybdenum disilicide coating (Figure 17). S/N 0005 engine was tested to the test matrix and accumulated a total of 4,613 seconds of burn time and approximately 20,500 starts.

S/N 0002 engine was tested and accumulated 592.8 seconds of burn time and 5,200 starts. It subsequently was subjected to launch vibration testing. Detailed inspection following vibration testing showed no adverse affects. Prior to post calibration tests, it was discovered that the exit nozzle had a crack at the edge of the bell section. Cause was attributed to mishandling.

Engine S/N 0003 was subjected to additional prototype tests including heat transfer tests, portions of the Mission Duty Cycle and steady state and pulse tests at off design conditions. During a series of tests to investigate the phenomena associated with combustion of vaporous rather than liquid N_2O_4, the chamber was subjected to an over temperature condition for an extended period



NEG. T3106-36

Engine S/N 0004 Combustion Chamber Burn Through

Figure 17

off time and burned through. The engine had accumulated a total of 874 seconds of burn time and 565 starts.

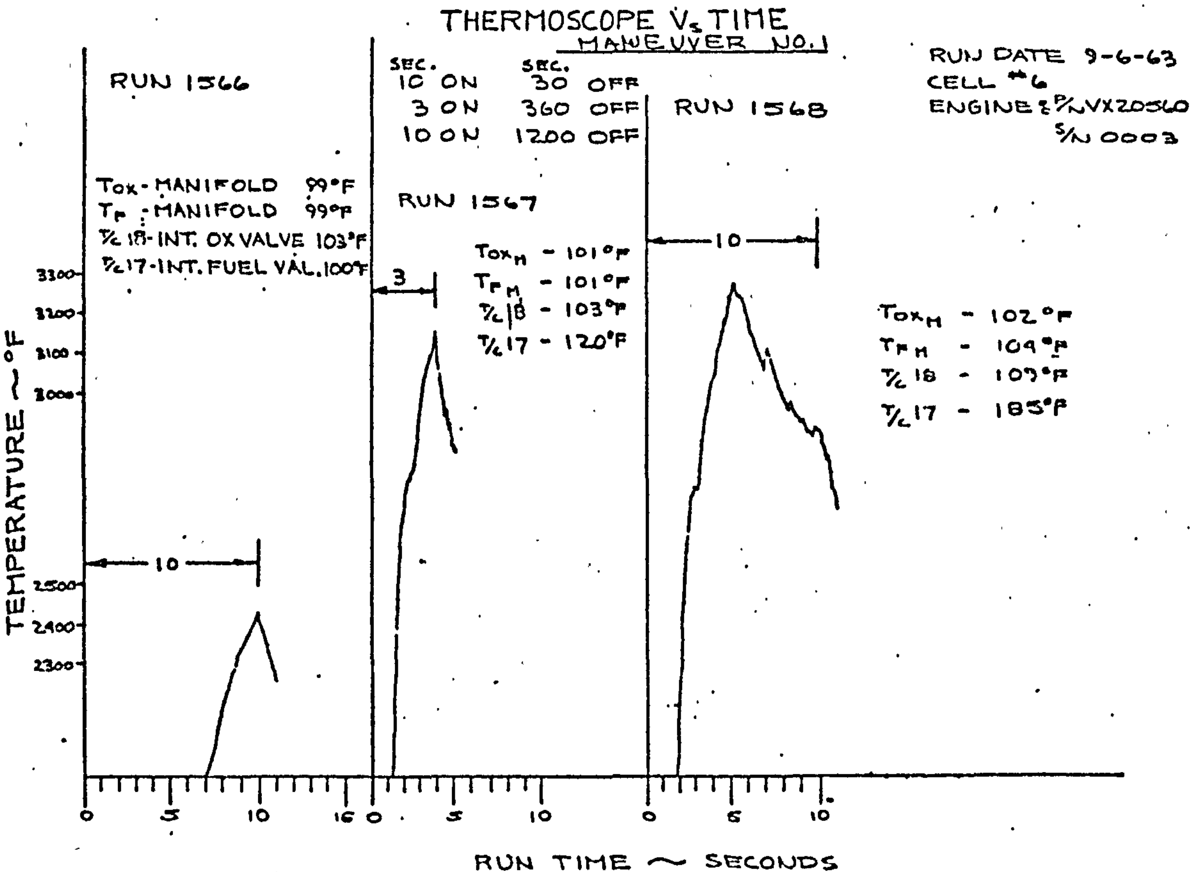
S/N 0001 engine successfully accomplished a firing test after its temperature had been lowered to -50°F. This engine was then subjected to a series of tests which include vaporization of the N_2O_4 and this chamber also burned through due to extended over temperature conditions. The engine had accumulated 110 seconds of burn time and 61 starts.

S/N 0006 and S/N 0007 engines were modified to include a design change in an attempt to alleviate the vaporized oxidizer hot starts. S/N 0006 was subjected to the tests and showed some improvement, however, the improvement was not completely satisfactory. S/N 0007 was tested in ATL Pad G. Starts of 11, 10 and 5 seconds were made. During a planned 5 second run, the thrust recorder indicated a slight rise in thrust, then a fall off. Investigation revealed that the combustion chamber had shattered.

As a result of the above, all prototype testing was halted and comprehensive investigations conducted into the causes of the above burn throughs and chamber failures. Tests with plastic chambers, utilizing high speed photography to study ignition processes were made as well as tests utilizing high speed oscillographs and oscilloscopes with special transducers to record pressure spikes.

The oxidizer vaporization phase stemmed from the Mission Duty Cycle Maneuver No. 1 (LM Transportation and Docking) which consisted of 10 seconds on, 30 seconds off and 10 seconds on. This vaporization phase or High Heat Transfer phase as it became known, is a phenomenon which had been noticed whenever a series of short steady state runs were made with short off times between runs. Under these conditions, it was noted that the combustion chamber wall temperature would exceed 3400°F for several seconds immediately following a restart-up in a series of runs and would remain in this "white hot" condition for a period of several seconds before returning to normal wall operating temperature of approximately 2800°F. See Figure 18 for a typical trace. It was theorized that after several short runs, the injector head had heated up to a value such that the oxidizer in the head and possibly even the solenoid valve, had vaporized and the elevated head temperatures continued to vaporize the oxidizer for a finite period of time. The resultant momentum of this vaporous oxidizer and the liquid fuel is such that the liquid fuel tends to flow down the core of the chamber leaving a vaporous reaction taking place at the chamber walls. This vaporous reaction is a more efficient combustion reaction than that of a propellant droplet reaction and thus generates a much higher heat release, overheating the chamber walls. As soon as liquid oxidizer again flows through the injector, conditions return to normal and the chamber wall cools down to its normal operating range.

COMBUSTION TEMPERATURE MANEUVER NO. 1



NOTE:
ALL T/C 18 & 17 TEMP. TAKEN AT END OF RUN.

Figure 18

A design review was conducted by NASA, NAA/S&ID and TMC on September 9, 10, 1963 covering these development problems.

The Prototype Development Program was stopped while detailed investigations were conducted into the causes for and the solutions to the combustion chamber burn throughs and fragmentations. The Prototype Development Program was rescheduled to start on or before 18 November 1963.

A comprehensive test program on the cause for and cures of combustion chamber fragmentation was conducted. This program utilized a direct connect test setup (see Figure 19) to evaluate both ignition phenomenon utilizing high speed motion picture cameras with plastic combustion chambers and combustion chamber ignition pressure spikes using two high response pressure transducers. The output of one transducer was recorded on a high speed oscillograph and the other was recorded on an oscilloscope. A camera was used to record the oscilloscope trace. During the tests, oxidizer and fuel leads were varied by means of an electrical delay system to the propellant solenoid valves. The leads were varied up to 20 milliseconds. Pressures obtained ranged up to 1100 psia and in general the two conditions which appear to contribute most to high ignition pressures were an oxidizer lead in to the combustion chamber and propellant temperatures at ambient or lower temperatures. One configuration change introduced and tested at this time was a "thirteenth " doublet located at the geometric center of the injector face. The purpose of the thirteenth doublet was an attempt to introduce propellants into the combustion chamber prior to the other doublets injecting and providing for preignition in an attempt to reduce ignition pressures. Data indicated some improvement.

Variations were made to the configuration to increase isolation of the oxidizer section from the injector head in an attempt to reduce High Heat Transfer phase burning. A steel isolation section between the oxidizer solenoid valve and the injector head was one approach. Tests indicated that the steel standoff was satisfactory except in the case where the propellant temperatures approached 100°F.

Controlled burst tests were conducted with sea level moly chambers with 0.042 inch thick walls by installing the moly chamber in a pressure vessel and pressurizing the outside of the chamber with water and the inside of the chamber with nitrogen. A burst diaphragm was installed on the water side of the test setup. Figure 20 presents a schematic of this test setup. Both pressures were then increased to the point where the burst diaphragm failed which imposed a rapid pressure decay on the outside of the chamber. Although this test did not accurately simulate the actual effect of ignition pressure buildup on the combustion chamber, it did provide a controlled means of applying a rapid pressure transient on the combustion chamber. Pressure decay rates of 1000 psi in 1.5 milliseconds were experienced during these tests,

PLASTIC CHAMBER TEST SET-UP

AEROTHERMO LAB

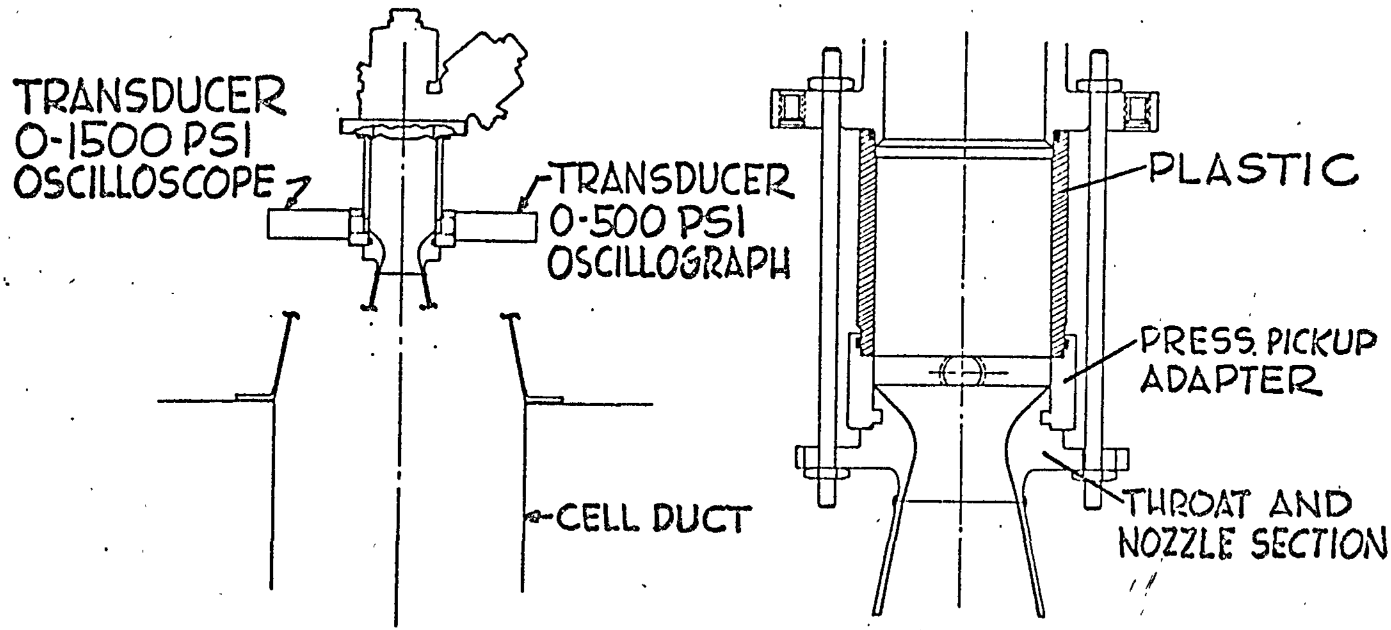


Figure 19

COMBUSTION CHAMBER

BURST PRESSURE TEST SETUP

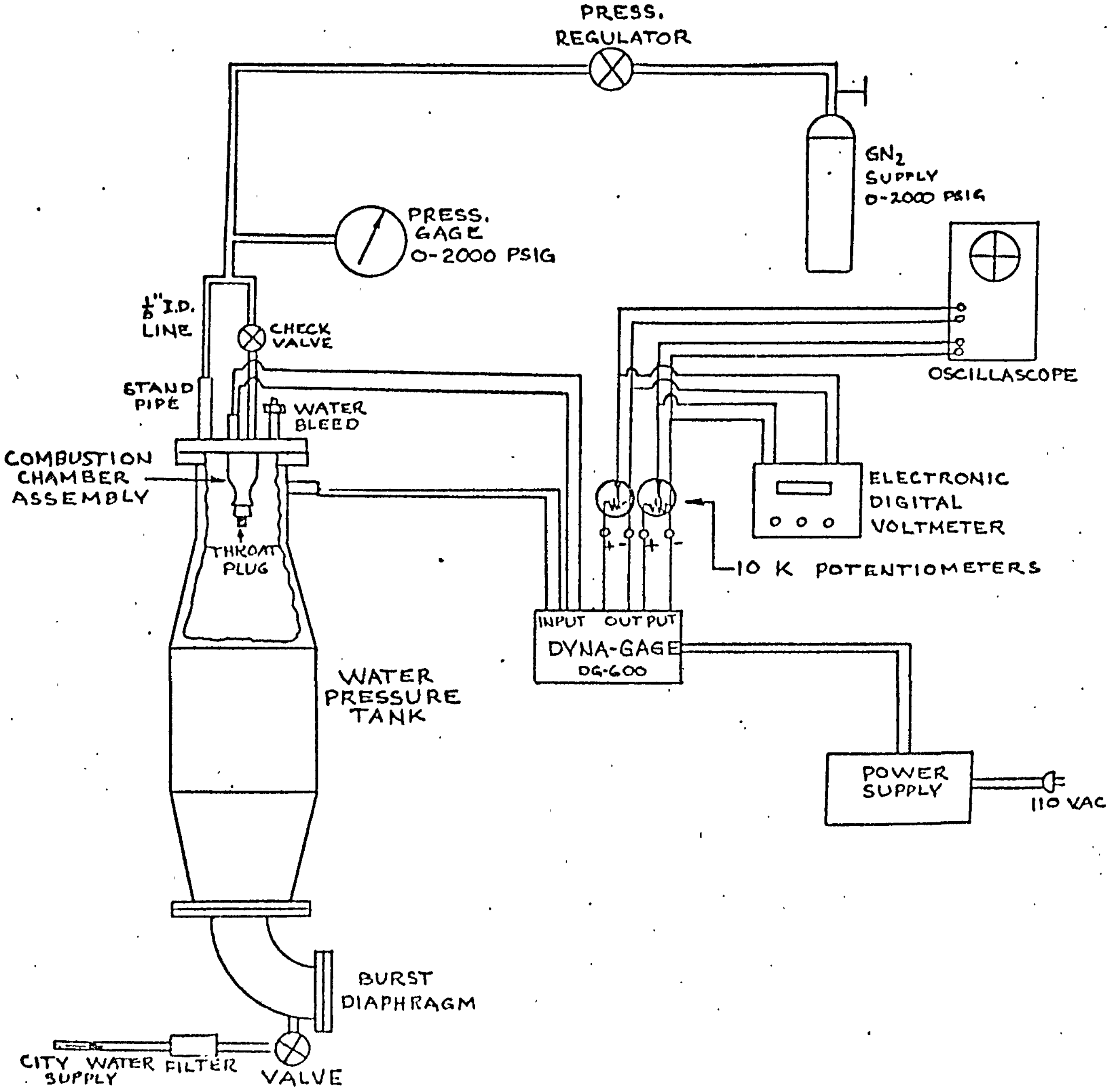


Figure 20

compared to ignition pressure transients of approximately 0.3 milliseconds. The results of these burst tests indicated the chamber failed between 1500 and 2000 psia.

C. X20560 Injectors

A total of ten injector configurations were designed during this phase of the program. The various injectors designed and tested are described below, including a description of the configuration change, the test results of the configuration, and a sketch of the configuration. Figure 21 presents a tabular summary of the injector configurations tested. Figure 22 presents a definition of terms used in the injector program.

Injector Configuration X20560-1 - This configuration incorporated a steel standoff section between the oxidizer solenoid valve and the injector head (Figure 23) supported by a phenolic adaptor. This standoff section was designed to reduce the maximum heat soakback temperatures of the oxidizer valve to 150°F. No improvement in pressure peak levels as compared to the X19900 engine tests was achieved. The engine high heat transfer operating condition was improved in that the maximum temperature the combustion chamber wall reached was approximately 3200°F and this condition lasted for only 5 seconds.

Injector Configuration X20560-501 - This configuration is the same as the -1 assembly except that the standoff has an I.D. of 0.090" oxidizer (see Figure 24). This configuration was designed to prevent expansion of the oxidizer to gas in the standoff section during engine operation at maximum heat soakback conditions. This configuration was intended to investigate heat transfer phenomena only. The 0.090" I.D. flow passage apparently aggravated the high heat transfer problem.

Injector Configuration X20560-503 - In this configuration, aluminum tube liners were inserted in the oxidizer cross passages (Figure 25) in an attempt to insulate the oxidizer flow from the injector head temperature conditions to prevent boiling of the oxidizer, a condition which contributes significantly to the high heat transfer burning phase. These aluminum tubes did not significantly improve the high heat transfer burning phase engine operating conditions and the configuration was dropped.

Injector Configuration X20560-505 - The diameter of the oxidizer cross feed passages was reduced from 0.043" to 0.031" in order to investigate the results of an analysis which showed that reduction of these passage diameters would be effective in preventing oxidizer boiling since the wetted areas of the oxidizer was reduced. Also, since it was desirable to have a fuel lead into the combustion chamber to alleviate the hard start condition, the injector insert was rotated 15° in the head so that one of the fuel holes was in line with the fuel inlet passage. This fuel hole is then fed by a dynamic pressure, thus

X20560 TEST ENGINE CONFIGURATIONS

FIG. NO.	DASH NO.	CONFIGURATION					PROPOSED SOLUTION FOR ENGINE PROBLEM OF:		REMARKS
		STANDOFF I.D.	TYPE OXIDIZER CROSS PASSAGES	HOUSING TYPE	DRIBBLE VOLUME				
					FUEL %	OXIDIZER %	P_c SPIKE	CHAMBER BURNTHROUGH	
II-3	-1	0.171	0.043 DRILLED	OLD PROTOTYPE	74	81	X	X	FIRST OF PART No.- BASE FOR OTHER VARIATIONS
II-4	-503	0.171	0.043 LINED WITH AL TUBES	OLD PROTOTYPE	74	77.6		X	INSULATED OX. PASSAGES TO PREVENT OX. BOILING IN THE HEAD ASSY.
II-5	-505	0.171	0.031 DRILLED	OLD PROTOTYPE - EXIT CUT NARROWED	64	67	X	X	REDUCED CONSTANT X-SECTION FUEL ANNULUS
II-6	-515	0.201	SAME AS FIG. 2	SAME AS FIG. 2	64	76	X	X	DOWNSTREAM OXIDIZER RESTRICTOR - 0.104 I.D.
II-7	-507	0.171	0.031 DRILLED	OLD PROTOTYPE - EXIT CUT NARROWED	66	67	X	X	13TH DOUBLET - REDUCED CONSTANT X-SECTION ANNULUS
II-8	-509	0.201	SAME AS FIG. 5	SAME AS FIG. 5	64	76	X	X	DOWNSTREAM OXIDIZER RESTRICTOR 0.104 I.D.
II-9	-511	0.171	0.031 DRILL	OLD PROTOTYPE - EXIT CUT NARROWED	58	68	X	X	FLAT-FACED INJECTOR - REORIENTED INSERT - TAPERED FUEL ANNULUS
II-10	-517	0.201	SAME AS FIG. 7	SAME AS FIG. 7	58	77	X	X	DOWNSTREAM OXIDIZER RESTRICTOR 0.104 I.D.
II-11	-501	0.090	0.043 DRILLED	OLD PROTOTYPE	74	55		X	(X20560 S/N 0003)
II-12	-513	0.090	0.031 DRILLED	OLD PROTOTYPE - EXIT CUT NARROWED	58	41		X	
	—	0.196	0.043 DRILLED	OLD PROTOTYPE	74	87	X	X	OLD PROTOTYPE - NO STEEL STANDOFF

Figure 21

DEFINITION OF TERMS

1. Old Prototype - Design used in X19900 engine (12-on-12) some of which have completed prototype testing, and incorporates the present steel standoff.

2. Standoff - The stainless steel adaptor which transports the oxidizer from the ox. solenoid valve to the injector head assembly and serves to isolate the oxidizer valve from the high head temperature conditions.

3. Oxidizer Cross Passages - Passages in the injector head assembly which transport the ox. discharge from the standoff to the injector holes.

4. Dribble Volume - Volume of the injector passages downstream of the solenoid valve poppet.

5. 100% Dribble Volume - Volume necessary to contain the propellant weight flow for one 10 millisecond pulse at an O/F ratio of 2.0 and a total flow rate of 0.333 pps.

6. P_c Spike - Short duration, high level combustion chamber pressures obtained upon propellant ignition.

7. Downstream Oxidizer Restrictor - Fixed orifice restrictor located downstream of the oxidizer solenoid valve.

8. Doublet - One oxidizer hole aligned with one fuel hole such that the streams issuing from same impinge upon one another.

9. 13th Doublet - As impiled, one doublet over and above previous twelve, located in the center of the injector face, fuel and oxidizer fed by dynamic pressure.

10. Reoriented Insert - Injector rotated 15° in the injector head assembly such that one of the fuel holes is aligned with the fuel inlet passage. The diameter of this fuel hole is o.018", while the diameter of the other eleven is 0.024". The 0.018 fuel hole is fed by a dynamic pressure, and sized such that it flows the same weight flow rate as the other eleven.

11. Flat-faced Injector - As implied - injector insert has a flat face, i.e., no metal cut away around the ends of the orifices. Provides very short free stream travel to impingement.

12. Tapered Fuel Annulus - Fuel annulus that is cut around the injector insert with an eccentric machining operation, such that the annulus cross-sectional area tapers from a maximum on the fuel inlet side to a minimum on the opposite side.

X20560-1 INJECTOR

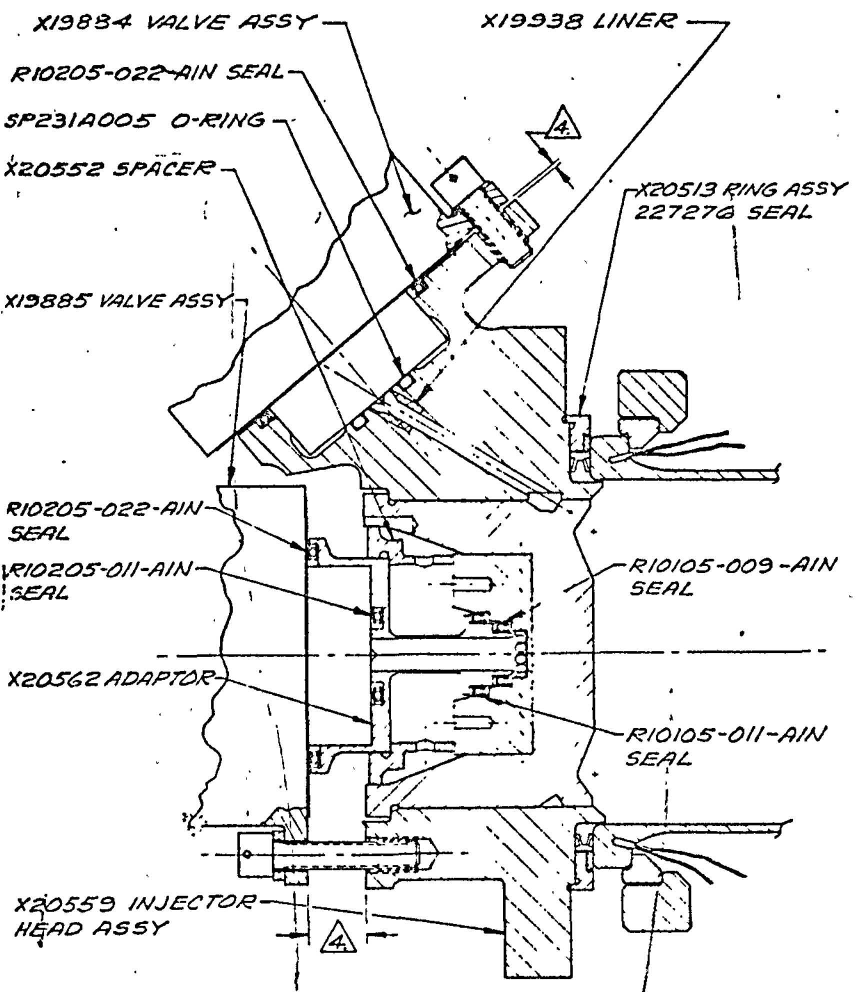


Figure 23

X20560-501 INJECTOR

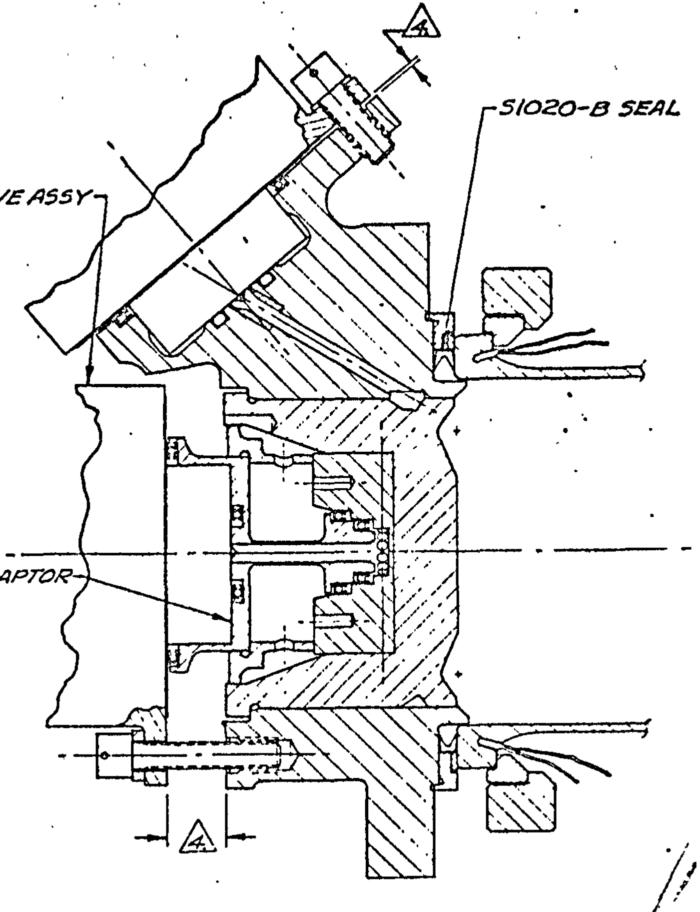


Figure 24

X20560-503 INJECTOR

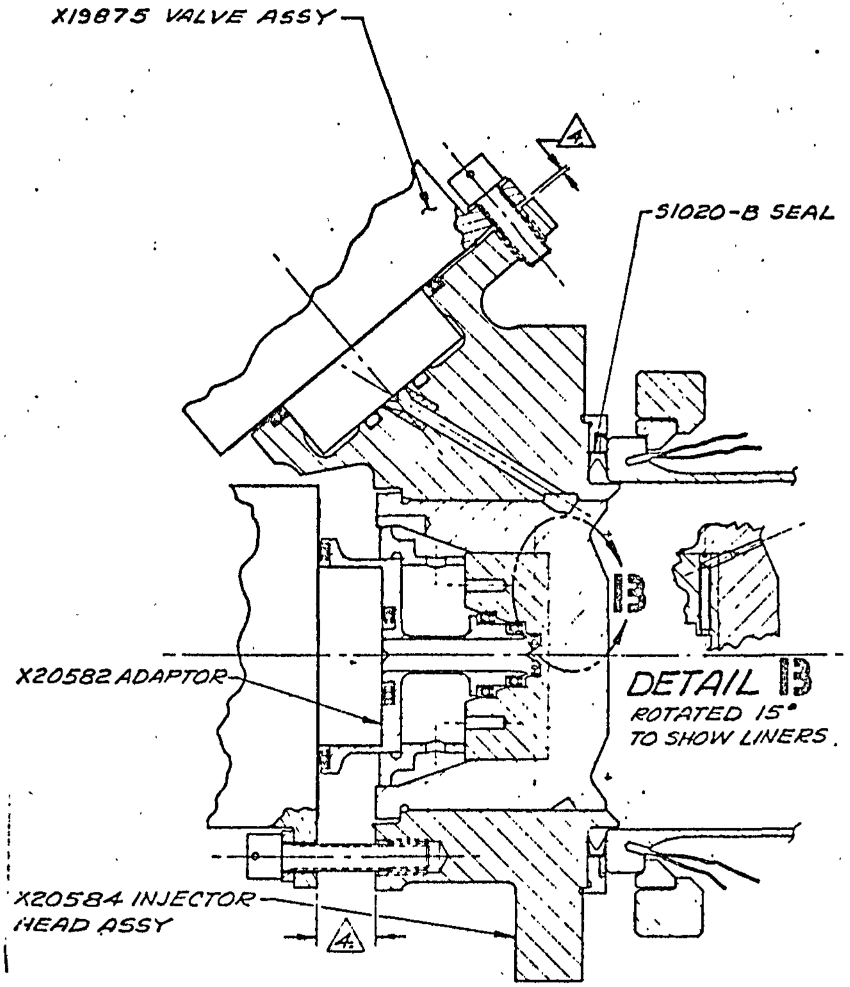


Figure 25

greatly decreasing the time from fuel valve open to fuel entry into the combustion chamber and creating a fuel lead. The fuel dribble volume was reduced from 74% in the -1 and -503 configurations to 64% in this configuration by reducing the cross sectional area of the fuel annulus in the injector head. The 0.031" oxidizer cross flow passaged reduced the oxidizer dribble volume to 67% (see Figure 26).

The engine high heat transfer burning phase was of approximately the same duration and severity for the 0.031" oxidizer cross flow passages as for the 0.043" passages. The reorientation of the fuel hole provided some slight improvement in the level of the ignition pressure spikes, but was not of a sufficient magnitude to be considered the solution to the problem. No effect on performance of the reduced dribble volumes was noted.

Injector Configuration X20560-507 - This configuration incorporated a 0.0171" width I.D. oxidizer standoff flow passage and 0.031" oxidizer cross passages. A flat-faced 13th doublet was added in the center of the injector face (Figure 27). Since the flow paths from the propellant valves to this 13th doublet were much shorter and of much less volume than the flow passages to the other 12 doublets, the propellants from this 13th doublet reached the combustion chamber in a much shorter time than from the other 12 doublets. Ignition of combustion at this doublet raises the chamber pressure, thus cutting down ignition delay at the other doublets. This faster ignition reduces the unburned propellant accumulation in the combustion chamber and contributes significantly to lowering the ignition pressure peak levels. Recent engine testing with varying cell pressure levels indicated that no damaging ignition pressure peaks are encountered if the chamber pressure is above 4 psia. Hence, the desirability of starting a localized reaction which increases the chamber pressure prior to the advent of the bulk of propellants into the combustion chamber.

The 13th doublet made some improvement in the ignition pressure spike level. Heat transfer tests were not conducted with this configuration.

Injector Configuration X20560-509 - This configuration was the same as the -507 assembly above except that the oxidizer standoff had a 0.201" I.D. flow passage and a fixed restrictor (Figure 28). The technical reasons for including the restrictor were the same as for the -515 assembly. Only ignition pressure spike data were obtained with this configuration. Since the preliminary examination of the data indicated no improvement in ignition pressure spike levels over that of the -507 assembly, data analysis on this configuration was stopped.

Injector Configuration X20560-511 - This was the same general configuration as the -505 assembly except that the injector face was flat to provide the minimum obtainable propellant free stream travel to impingement, and

X20560-505 INJECTOR

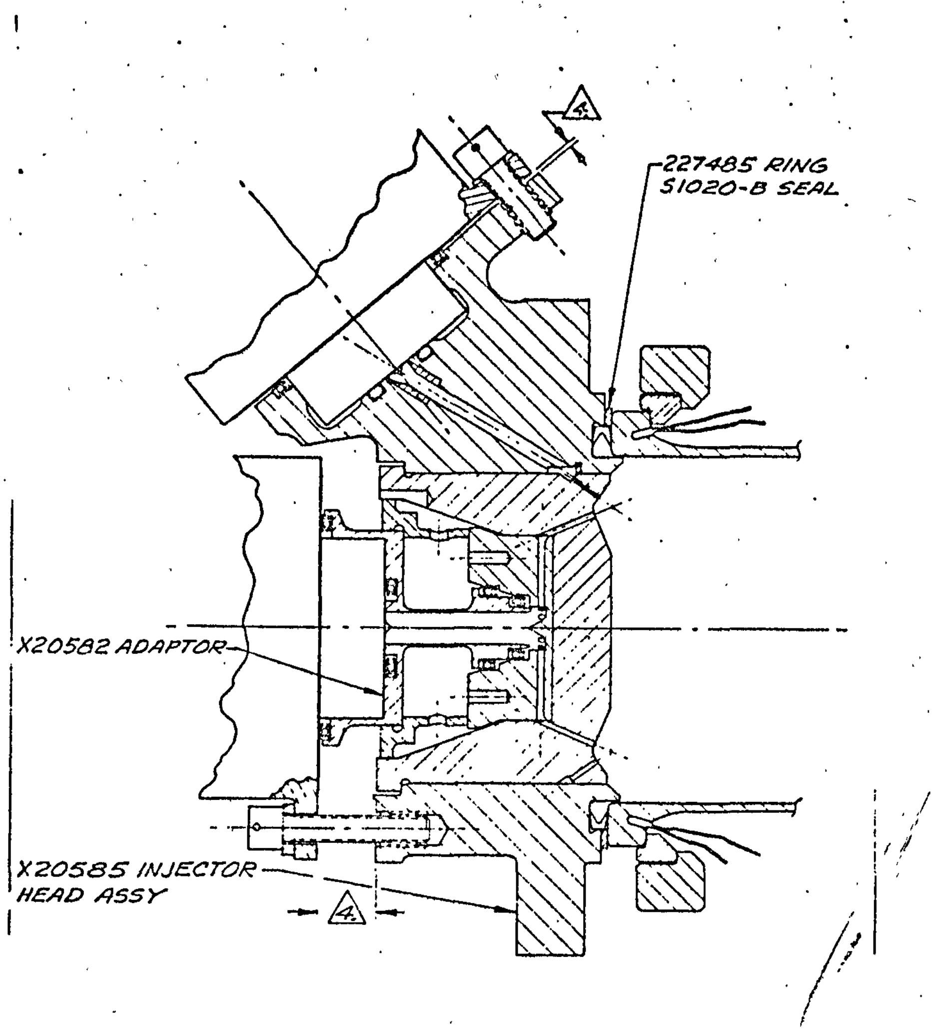


Figure 26

X20560-507 INJECTOR

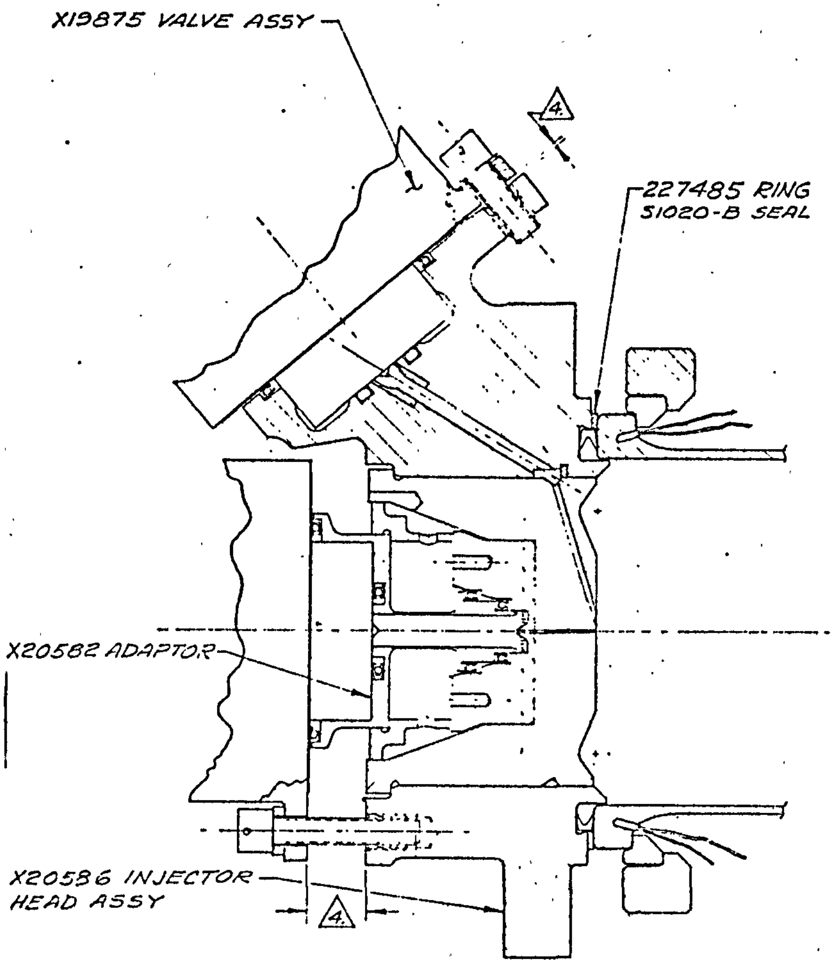


Figure 27

X20560-509 INJECTOR

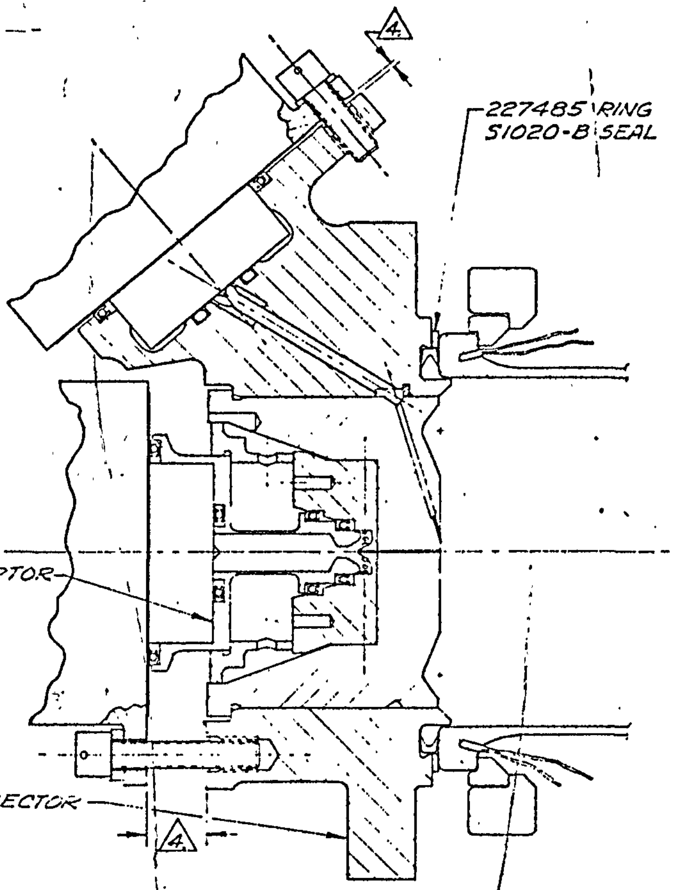


Figure 28

the fuel dribble volume was reduced from 64 to 58% because of a tapered fuel annulus passage (Figure 29). Containment of the liquid streams closer to the impingement point prevents their expansion to the gaseous state and the consequent freezing of part of the oxidizer, the latter condition contributing to ignition delay. At the same time, liquid phase mixing is promoted which is essential to good ignition characteristics. The reoriented insert (one fuel hole in line with the fuel inlet) and the smaller fuel dribble volume ensure a fuel lead into the chamber.

The ignition pressure spike characteristics were much better with this configuration. This is probably primarily due to the shorter free stream propellant travel of the flat face configuration, but may also be influenced to a degree by the smaller fuel dribble volume. High heat transfer burning phase operation was also reduced to an acceptable level.

Injector Configuration X20560-513 - This is the same configuration as the -501 assembly except that the oxidizer cross flow passages are 0.031" diameter rather than 0.043" (Figure 30), in order to alleviate a second flow expansion area in the injector head and thus further inhibit the tendency of the oxidizer to gasify at high heat soakback conditions.

Injector Configuration X20560-515 - This injector had the same configuration as the -505 injector except that the standoff had an oxidizer flow passage I.D. of 0.201" rather than the 0.171" in the -505 assembly and incorporated a fixed area restrictor at the discharge end of the passage (Figure 31). As temperatures increase and pressures decrease, most of the N_2O_4 oxidizer tends to vaporize upon exiting from the valve into a relatively large cavity. Calculations indicated that a fixed restrictor in the standoff would cause the oxidizer flow to choke at that point, thereby causing the pressure in the standoff passage to rise and thus preventing further vaporization of the oxidizer in the standoff. In addition, this restrictor would tend to delay the arrival of the oxidizer into the combustion chamber by approximately one millisecond and thus further assure a fuel lead. This downstream restrictor provided no significant improvement in the ignition pressure spikes so no further testing was conducted with this configuration. No heat transfer data were taken with this configuration.

Injector Configuration X20560-517 - The -517 injector was identical to the -511 flat face, rotated insert configuration except for the substitution of the 0.201" I.D. oxidizer standoff, with downstream fixed restrictor. Fuel dribble volume was 58% and oxidizer dribble volume was 77% (See Figure 32).

As in the -515 configuration listed above, the design was to do two things: (a) reduce or eliminate possible vaporization of the oxidizer in the oxidizer standoff area, and (b) by rotation of the insert insure a fuel lead into the combustion chamber.

X20560-511 INJECTOR

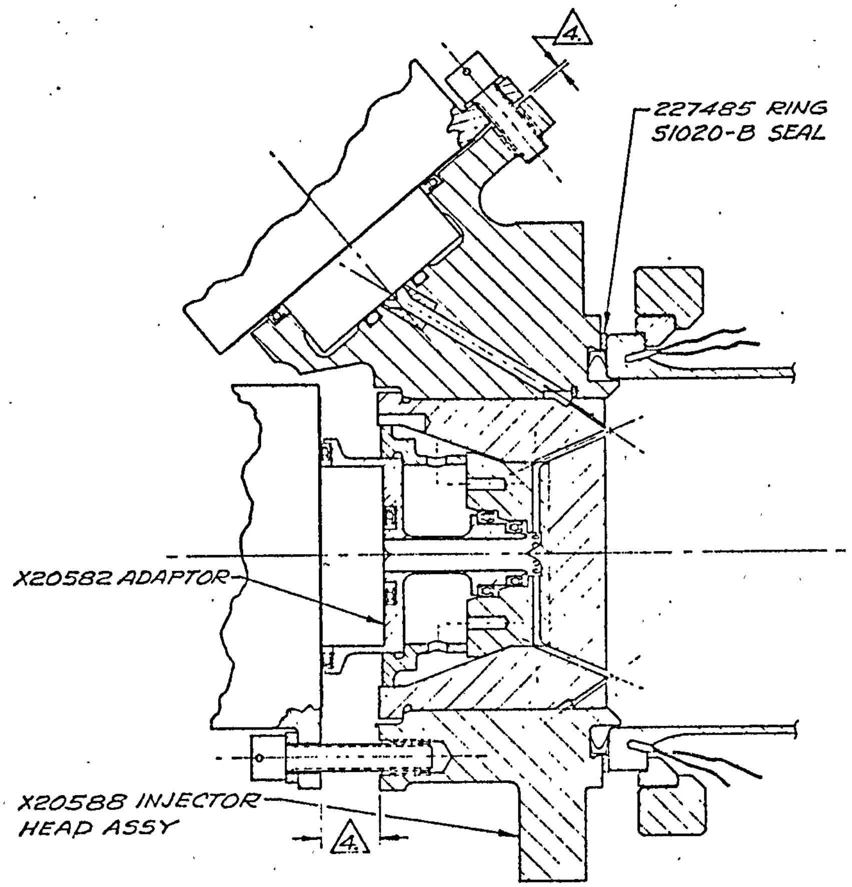


Figure 29

X20560-513 INJECTOR

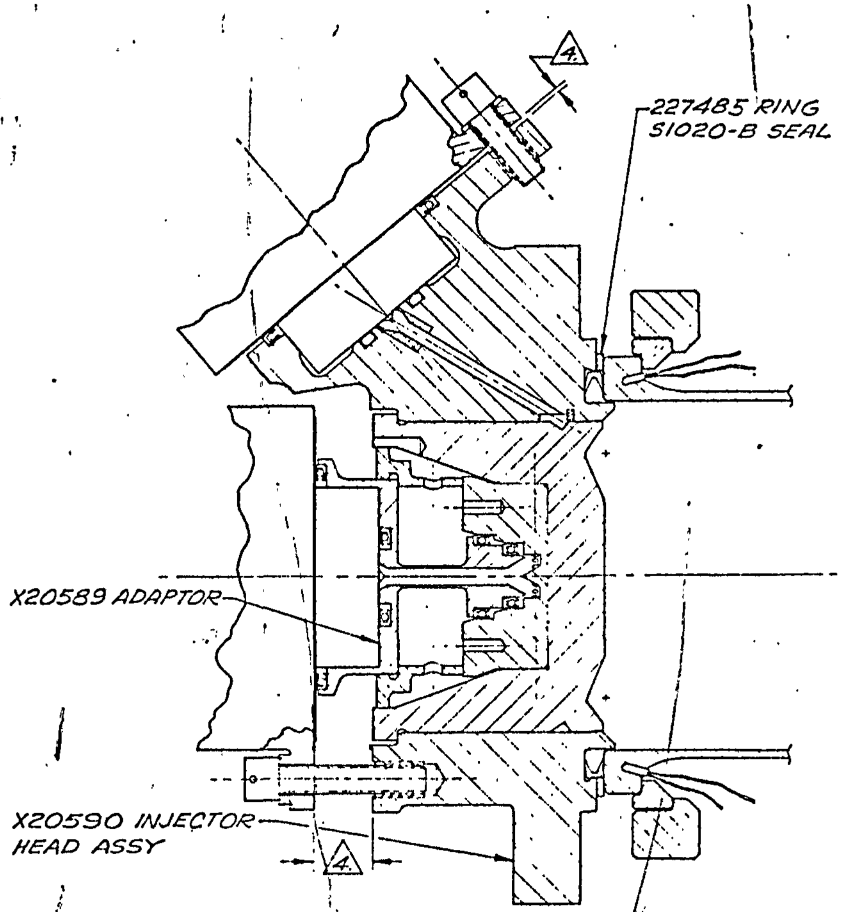


Figure 30

X20560-515 INJECTOR

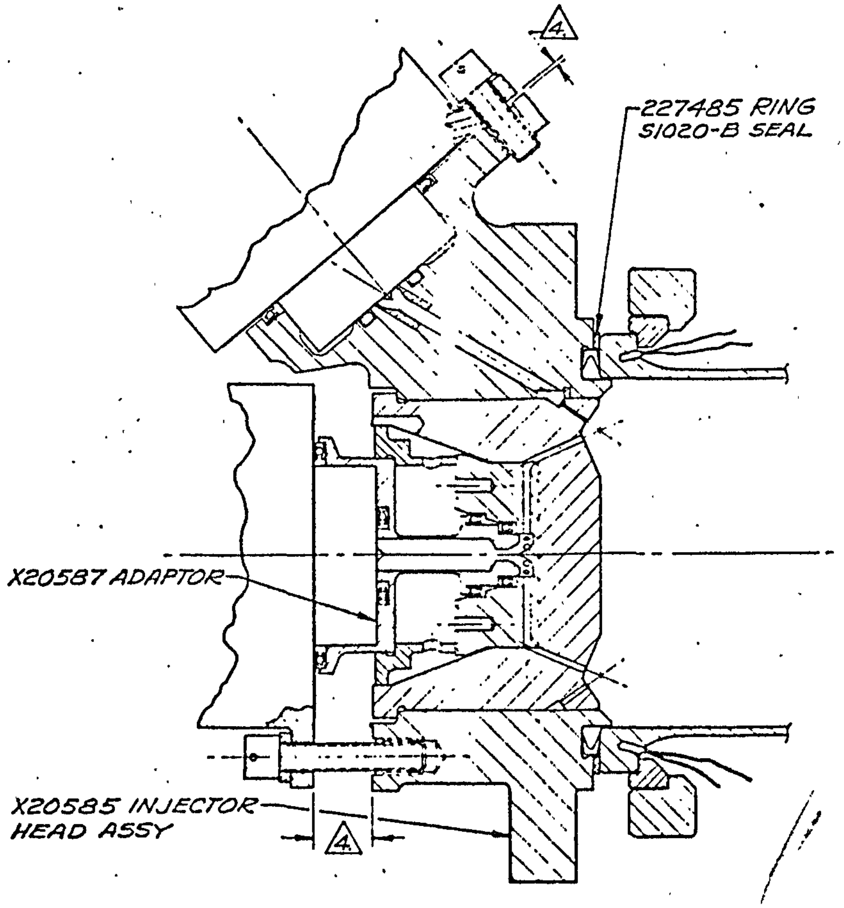


Figure 31

X20560-517 INJECTOR

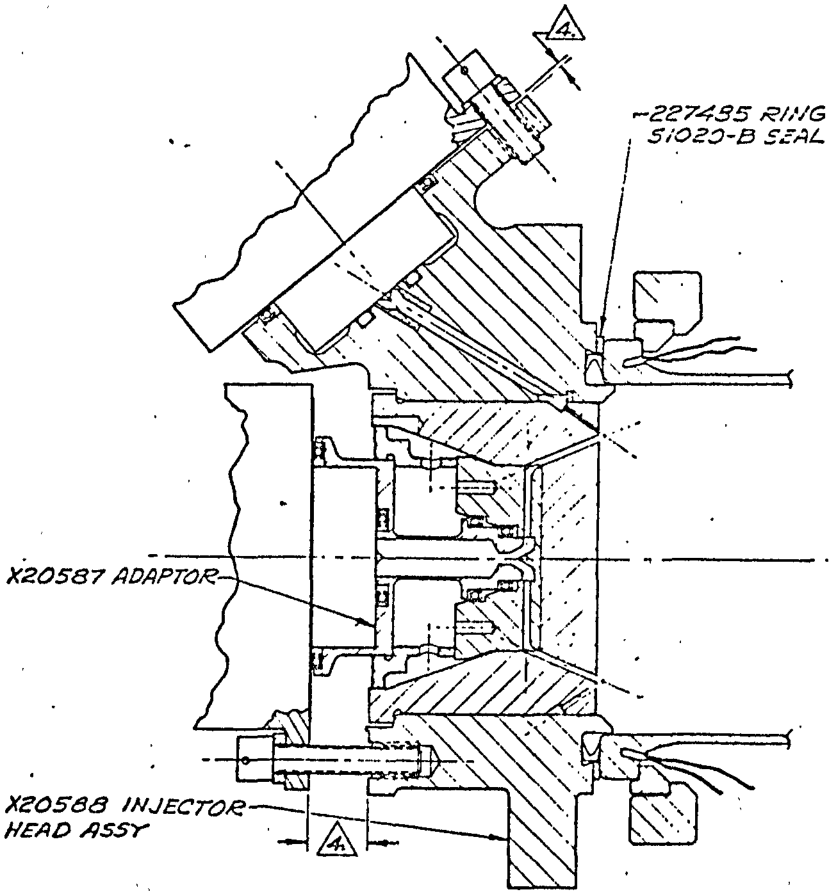


Figure 32

Test data indicated that the ignition pressure spike levels recorded for this engine were equivalent to those recorded for the -511 configuration.

No high heat transfer data was obtained for this configuration.

Injector Configuration X20560-519

The -519, 12 on 12 configuration utilized the 0.171 standoff, and a fuel inlet passage diameter of 0.073 inches. It was a flat-faced injector with a normal orientation insert (fuel inlet between #1 and #12 doublets) and utilized an eccentric fuel annulus. The fuel hole diameter was unchanged at 0.024" but the oxidizer hole size was reduced from 0.031 on previous injectors to 0.026". The momentum angle at an O/F = 2.0 was +5°. The fuel dribble volume was 58% and the oxidizer dribble volume was 80% (see Figure 33).

A flat-face was utilized in the -519 configuration to provide good ignition characteristics. The insert was installed in the normal orientation to provide uniform steady state wall heating. The eccentric fuel annulus was utilized to minimize the fuel dribble volume while providing equal fuel hole flows. The oxidizer hole size was reduced to test the effect of higher velocity oxidizer injection on ignition peak pressure levels and on steady state wall heating. The momentum angle of +5° at O/F = 2.0 was to provide increased wall cooling which appeared necessary since previous high injection velocity engines tended to have hot walls.

The -519 configuration did not receive extensive high heat transfer testing. Only one 4 second run was made in Cell 6. Chamber wall temperature exceeded 3100°F at this time and the test was discontinued.

Injector Configuration X20560-521

The -521 injector was an 8 on 8 flat face insert (normal orientation), $D_{ox} = .029$, $D_f = .029$, $D_{imp} = 700$, M.A. = +16°, $D.V._{ox} = 68.4\%$, $D.V._f = 67\%$ and A = 0.171 oxidizer standoff. The chamber was a standard moly non-grit blasted chamber with a standard K-seal with Rene' heat barrier spacer (see Figure 34).

The -521 engine was designed to verify previous performance results with 8 on 8 injectors using the same hole sizes and included stream angles but incorporating the new oxidizer valve "standoff" concept to prevent high heat soakback temperatures to the oxidizer solenoid valve. All previous similar engines had run cool but had not been tested for high heat transfer burning operation. It was reasoned that the low engine temperatures would not bring the propellants to a temperature which would promote high heat transfer burning and a flat face was incorporated in the injector to aid the ignition process.

X20560-519 INJECTOR

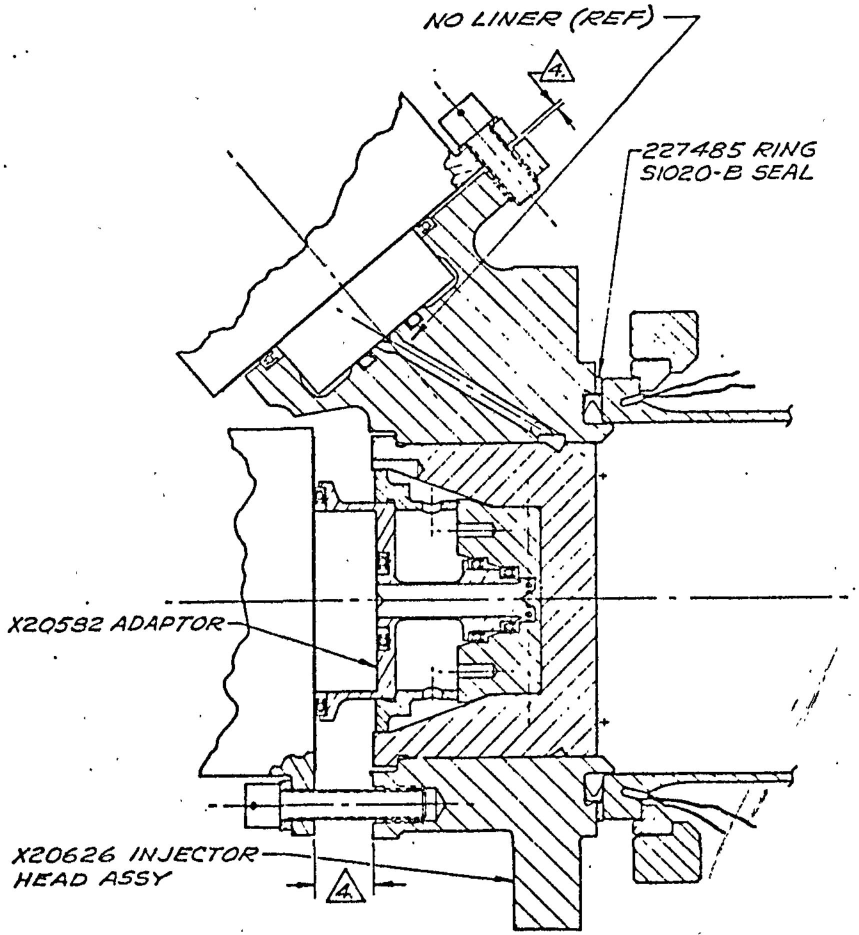


Figure 33

X20560-521 INJECTOR

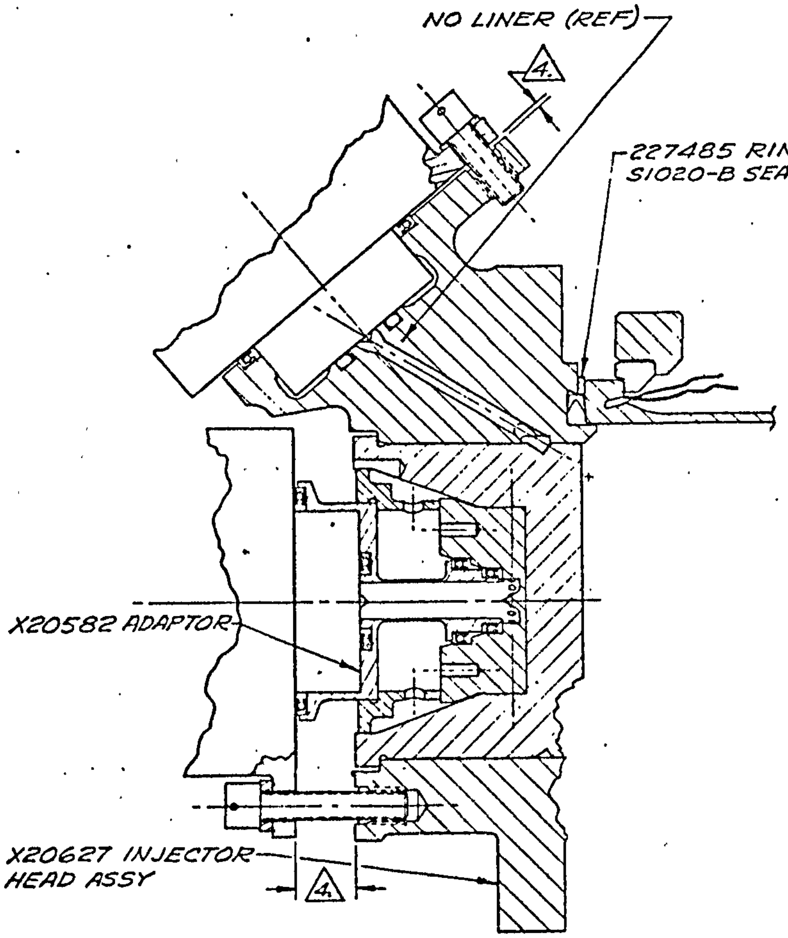


Figure 34

High heat transfer phase burning was greatly improved but not eliminated. The chamber wall temperature was extremely low (2000-2245°F) as well as the moly flange temperatures (1227-1430°F) for 60 second steady state runs at O/F = 2.0. Steady state I_{sp} was 280 seconds for an O/F = 2.0.

Other configurations of the -521 were tested. These are described below:

The second configuration was the same as above but with a standard moly grit-blasted chamber and a wide "K" seal. This variation was conceived to increase thermal resistance between the chamber and head, thereby reducing heat transfer back to the head and eliminating high heat transfer phase burning. It was also designed to decrease chamber wall temperature. No detrimental high heat transfer burning was experienced at O/F = 2.0 at propellant temperatures from 100-118°F. The chamber wall temperature was extremely low (1990-2235°F) as well as the moly flange temperature (1447-1489°F). The steady state I_{sp} was 276 seconds for an O/F = 2.0. The pulse I_{sp} was within specification requirements up to 50 ms pulse widths.

The third configuration was the same as the first -521 but with a standard moly grit blasted chamber and a lucalox (ceramic) seal between the injector head and the combustion chamber. This change was made to increase thermal resistance between the chamber and head, and therefore to eliminate high heat transfer burning. It was also run to evaluate the lucalox seal. The injector head temperature was approximately 50°F higher than previous tests as the lucalox seal did not conform to thermal resistance predictions. High heat transfer phase burning was experienced during extended maneuver #1 because of higher head temperatures caused by the lucalox seal. The chamber wall temperatures were low (1900-2100°F) as well as moly flange temperatures (1135°F) for 60 second steady state runs at O/F = 2.0. The steady state I_{sp} was 280 seconds for an O/F = 2.0.

A fourth configuration of the -521 injector was tested. This configuration was the same as the first -521 except for a ribbed moly sleeve with strain gages which replaced the standard combustion chamber. It was instrumented with 140 kc Kistler transducers. The tests were run on this configuration to document ignition characteristics. The maximum pressure recorded (from strain gages) was 1720 psi This is an improvement of approximately 27 percent over the 12 on 12 engine at 2490 psi.

Injector Configuration X20560-523

Free stream travel of the injected streams was minimized in this configuration and the injector face was "vee" cut to inhibit fuel to oxidizer interpassage flow, resulting in the "knobby" face configuration.

The configuration used a normally oriented injector insert. An eccentric (tapered) fuel annulus was incorporated. The injector hole sizes and angles are the same as the -1.

Short free stream length was used to capitalize on the flat-faced (-511) type of injectors lower ignition peak pressures, while the incorporation of the "knobby" face design prevented interpassage flow of the propellants. The eccentric fuel annulus further reduced the fuel dribble volume to ensure a fuel lead into the chamber to further minimize ignition pressure peaks.

The normally oriented injector insert was designed to minimize steady state wall temperature variations.

The anticipated steady state performance would have been identical to the -1 configuration because of the injection similarities. Consequently, no high heat transfer tests were performed on this configuration. Ignition peak pressure tests indicated peak ignition pressure levels comparable to those measured with the -511 configuration.

Injector Configuration X20560-525

The -525, as did the -523 configuration, incorporated the "knobby" face and minimum free stream length design. The Number 1 fuel orifice of the injector insert was rotated 15° to line up with the fuel inlet. The angle of the Number 1 oxidizer hole was increased and its inlet location changed to provide it with a total pressure source. The fuel hole sizes were the same as for the -511 configuration. The Number 1 oxidizer hole diameter was .0352 inches and the other eleven were .0314 diameter.

The knobby face, designed for minimum free stream length, prevented explosions in the manifold due to propellant flow from one hole to another. Rotation of the injector insert, and increased angle of the Number 1 oxidizer hole, were design changes to assure early propellant arrival at the Number 1 doublet, thus raising chamber pressure prior to flow from the remainder of the doublets and thereby lowering the ignition peak pressure.

In addition, the increased angle and total pressure source for the Number 1 oxidizer hole were designed to counteract the high momentum of the Number 1 fuel stream providing a more uniform spray pattern than the similar -511 configuration.

Ignition peak pressures measured on the -525 were comparable to those of the similar free stream to impingement length -511 configuration. The -525 injector insert face also was better than the -511 in that no deterioration of the injector hole exits was noted.

The configuration was subjected to a limited heat transfer test series in Cell 1 and showed only slight evidences of high heat transfer burning at O/F = 2.0.

Injector Configuration X20560-527

The -527 injector configuration is identical to the -519 configuration with the exceptions that the injection hole diameters are smaller, D_f = 0.021 and D_{OX} = 0.024, and the oxidizer cross passages are 0.043 instead of 0.031. These diameters create a momentum angle of +3.8° at an O/F = 2.0 with an oxidizer hole angle of 23° and a fuel hole angle of 35°. The fuel dribble volume is 57% and the oxidizer dribble volume is 80%.

The fuel and oxidizer holes were made smaller in the -527 injector to promote better propellant mixing for "softer" engine ignition by providing higher stream velocities and to inhibit the high heat transfer burning that the propagated through boiling of the oxidizer by keeping the propellants at a higher pressure level in the injector. The oxidizer and fuel hole diameters were changed to provide a +3.8" (outward) momentum angle to provide a reasonable chamber wall temperature since previous experience with high velocity streams indicated high wall temperatures. The injector insert was used in the normal orientation to provide a uniform wall temperature.

Two heat transfer test series were attempted but in both cases the engine suffered combustion chamber bell breakage (flow turned chamber) which stopped the testing. In the limited amount of heat transfer data which was obtained, the engine indicated high heat transfer burning when fired at maximum soakback conditions and showed no tendency to step out of the condition. As with the other flat faced injectors, such as -511, the injector hole exits exhibited erosion after approximately 627 seconds of burn time. Exit erosion appears to be worse on the smaller holes.

No ignition peak pressure tests were run on this configuration.

Injector Configuration X20560-529

The -529 injector was made the same as the -511 configuration with the following exceptions:

(a) A concentric fuel annulus providing 64% fuel dribble volume was used.

(b) An insulating air gap was left between the insert inner body and the insert main body. A small bleed hole was drilled connecting the air gap to the combustion chamber.

(c) The fuel dribble volume is 64% and the oxidizer dribble volume is 67%.

The -529 configuration was made with a concentric fuel annulus because it provides more nearly equal flow out of the fuel holes. The "air gap" was provided between the bottom of the inner body, which contains the oxidizer cross passages, and the insert main body, to minimize the heat transfer to the oxidizer and prevent it from boiling in the cross passages, which appears to cause high heat transfer burning. The small bleed hole was provided to the combustion chamber to prevent any gas formed from oxidizer trapped in the air gap from being forced into the oxidizer streams. A flat face was incorporated to minimize peak ignition pressures.

Limited heat transfer tests were performed on the -529 configuration in Cell 6. Combustion chamber wall temperatures from cold head starts were over 3100°F on two 60 second run attempts. On succeeding runs wall temperatures were approximately 2700-3000°F. The engine showed no improvement over the -511 configuration during high heat transfer burning. The combustion chamber was blistered during the test series due to excessive wall temperatures.

Injector Configuration X20560-531

The -531 engine was 0.5 inches longer than the other engines since it had straight, drilled oxidizer injection passages. The oxidizer passages were counterbored 0.044 diameter but the oxidizer and fuel injection hole diameters were the same as the -1 configuration. The oxidizer hole angle was increased from 23° to 31.5° which increased the momentum angle at an O/F = 2.0 from 0° to +5°. The fuel dribble volume was 58% and the oxidizer dribble volume was 80%.

Straight oxidizer passages were used in the -531 configuration because it was reasoned that the turns in the oxidizer passages on the previous injectors were inducing cavitation in the oxidizer streams, causing bubbles to form which in turn caused two phase (gas plus liquid) flow from the injector, thereby propagating the unstable combustion manifested in the high heat transfer burning phase. The oxidizer stream angle was changed from 23° to 31.5° to provide a +5° (outward) momentum angle to help cool the chamber wall with propellant spray. The injector had the flat face for good ignition characteristics.

The -531 configuration showed no significant improvement in high heat transfer burning. Steady state wall temperatures were approximately 2800°F on the only long steady state run performed on the engine. Tests were stopped because the combustion chamber had started to melt during an 8 second run following a 60 second soakback run.

Injector Configuration X20560-533

The -533 configuration was the same as the -531 configuration with the following exceptions:

(a) Steel insulating tubes with the same I.D. as the previous drilled injector holes were driven into the oxidizer passages.

(b) The steel tubes eliminated the oxidizer hole counter bores.

(c) The oxidizer dribble volume is 78%

The -533 configuration oxidizer passages (straight) were insulated by driving steel tubes into the passages which were supported at both ends, because the X20560-531 which was of the same design in all other respects, went into the high heat transfer burning phase due to overheating and boiling of the oxidizer in the injector passages. It was anticipated that this configuration would prevent the high heat transfer burning phase and make unnecessary the building of an injector with individual independent oxidizer injection tubes. The flat face was incorporated in the -533 configuration to provide good ignition characteristics.

The -533 configuration high heat transfer performance was not significantly better than the -531 or -511 configurations. The steady state chamber wall temperatures were somewhat lower, however, during a steady state run of 37 minutes to evaluate chamber life, chamber wall temperatures were approximately 2800°F and chamber flange temperatures stabilized at 1800-1850°F while the engine maintained a mean I_{sp} of 300 seconds.

Of the above configurations, X20560-511 had demonstrated compliance with the required duty cycle and satisfactory solution to the two problems.

Problem		Design Solution
Ignition Pressure Spikes	-	Hydraulic Fuel Lead
High Heat Transfer Burning	-	Isolated Oxidizer Standoff

D. Development Program

In mid-November 1963, effort was initiated toward achieving a better basic understanding of the failure mechanisms and to develop an engine with lower operating temperatures, greater structural margin and lower ignition transient pressures. In addition, a parallel program was initiated to develop an interim engine configuration which would require minimal changes to specification requirements. This was designated as "Specification Change Engine Program".

The following represented the Development Program in late November, 1963.

1. Applied Research Program
   - (a) HHTB Studies
   - (b) Ignition Pressure Spiking Studies
   - (c) Alternate Materials
2. Configuration Improvement/Engine Evaluation Program
   - (a) High Heat Transfer Burning Test Evaluation
   - (b) Ignition Pressure Spiking Test Evaluation
   - (c) Increased Engine Operating Life Test Evaluation

     (Additional Combustion Chamber Wall Temperature Margins)
3. Specification Change Engine Program

The Applied Research Program involved fundamental analysis and laboratory experimental investigations while the Configuration Improvement/Engine Evaluation Program covered the full scale testing necessary to evaluate and demonstrate the adequacy of the Applied Research Program design rationale. The Specification Change Engine Program involved analysis, data research and full scale testing of engines at O/F's of 1.3 and 1.6.

The High Heat Transfer Burning (HHTB) studies involved analysis and evaluation of engine data, postulate models and critical experiments including transparent chamber engine tests with both normal burning and HHT/Burning utilizing high speed photography. In addition, coating evaluation and emissivity improvement studies were undertaken. The Ignition Pressure Spiking Studies included analysis and critical experiments in relation to combustion kinetics, structural analysis and test of cylindrical and contoured (ribbed) moly sleeves including combustion tests, bending tests and Bermite explosive tests.

Full scale tests were accomplished with various configuration changes in order to evaluate potential improvements.

Results of these programs during late 1963 and early 1964 indicated that HHTB was a result, at least in part, of two phase flow of the oxidizer and is a mode of rough combustion with high random noise level at lower frequencies compared to normal burning. Emissivity studies of coated molybdenum tubes, machined and grit blasted showed:

1. Coating emittance increases slightly with increase in temperature for both machine and grit blasted surfaces.
2. Coating emittance is higher for grit blasted surfaces compared to machined surfaces.
3. Coating hemispherical total emittance changes are negligible with respect to time.
4. Coating spectral emittance increases slightly with respect to time.

Combustion kinetics studies continued, improved pressure transducer with smaller diaphragms and higher natural frequencies (fn > 120 kc) were used and improved data acquisition were developed and utilized in measuring ignition pressures. Pre-mix chambers were evaluated along with burst tests, bending tests and explosive charge tests. Tensile testing at -40 and +32°F were also conducted.

The X20560-521 engine configuration was further analyzed and tested as a Specification Change engine and engines T9661 (O/F 1.6, 12 on 12 injector), T9662 (O/F 1.6, 8 on 8 injector), T9663 (O/F 1.3, 12 on 12 injector) and T9664 (O/F 1.3, 8 on 8 injector) were in work.

During this time period, changes were made to the solenoid valves. The major difference in design was to increase the number of ampere-turns of the manual coil, which requires a large window winding area, eliminating the recess at the back of the armature and electrolizing the valve body internal diameter. Figure 35 presents the old and the new configurations.

At the same time, due to the increase in the test load, MJL Cell No. 1 was being activated into an altitude facility (previously a sea level facility).

SOLENOID VALVES

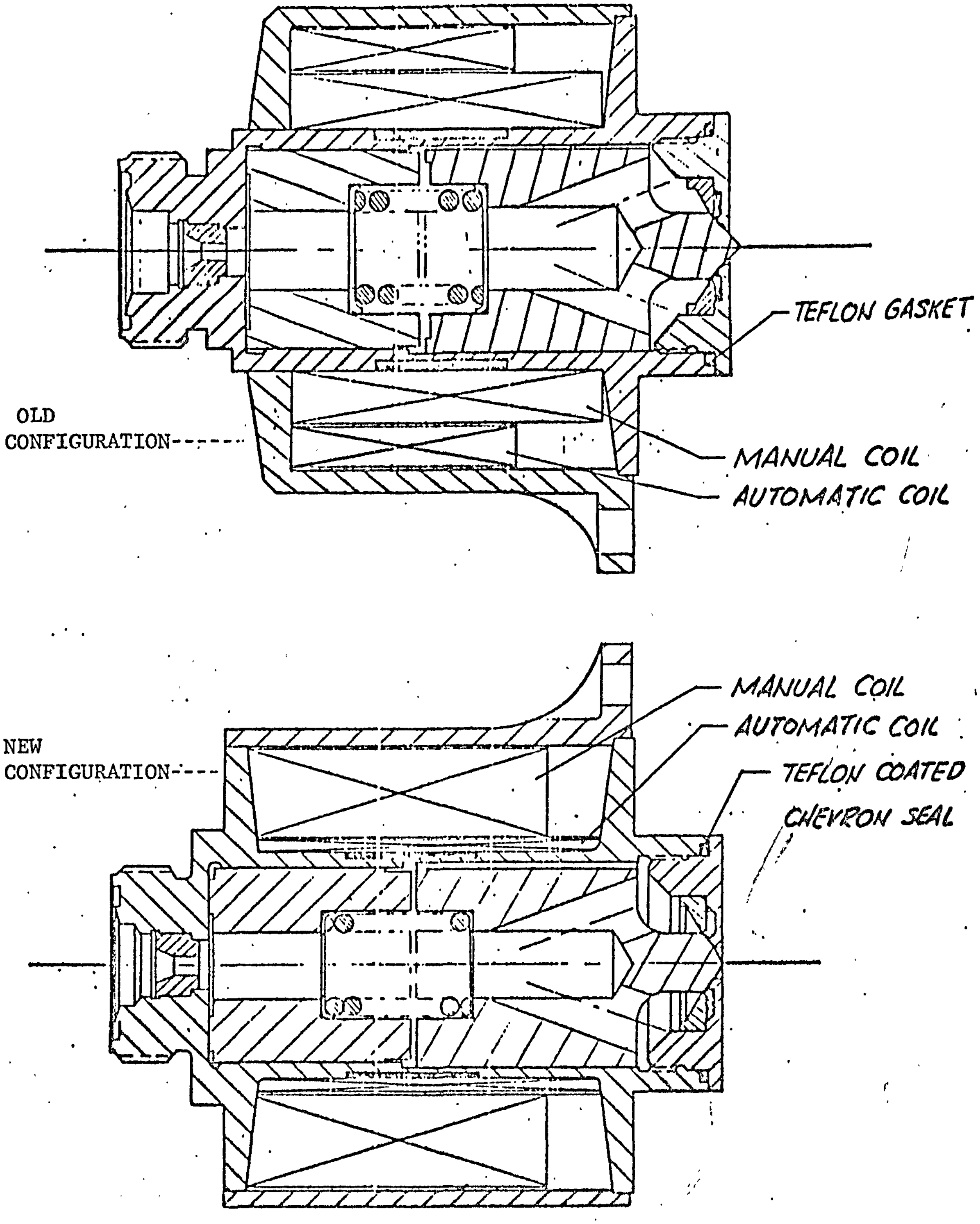


Figure 35

III. FUEL FILM COOLING

During early February, 1964, modifications were made to both the 8 on 8 and 12 on 12 configurations to include film cooling of the combustion chamber by use of fuel bleed holes. The T9730 engine with an 8 on 8 injector configuration was modified to include 8 bleed holes radially in line with the 8 main doublet pairs and at an injection angle into the combustion chamber approximately 25° such that the fuel impinged on the combustor wall approximately 0.5 inch from the combustor flange. (The percent of total fuel flow used for bleed in this configuration was 11%).

Results of these tests showed that at design conditions of O/F = 2.0, F_t = 100 lbs., the I_{sp} was 295 seconds, the chamber wall temperature was 2500°F and the chamber flange was 2000°F. This compares to performance of a like configuration T9662, without fuel film cooling, which had an approximate I_{sp} of 304 at an O/F of 2.0 with wall temperatures of 2900 to 3000°F and flange temperature of 2200 to 2300°F. Additional modifications of the T9730 engine are shown on Figure 36.

A 12 on 12 configuration was also modified to include 12 fuel cooling holes located radially out from the main pair of doublets. Impingement point was 0.09 inches below the injector face. This configuration was identified as T9715. This engine also demonstrated lower wall and flange temperatures, i.e., at an O/F of 1.6, I_{sp} = 285 seconds, wall temperatures equalled 2700°F and flange temperatures were 900°F. At an O/F of 2.0, I_{sp} 284 seconds and wall temperatures were at 2790°F. Maximum soakback to the injector head after a 60 second run was 200°F at both O/F = 1.6 and 2.0.

A schematic of Engine T9640 is shown on Figure 37. This configuration incorporated a premix chamber to provide for test evaluation to determine the capability of premix chambers to shorten ignition delay time and reduce the magnitude of ignition pressure spikes.

Effort continued during March, 1964, toward optimization of fuel cooling and reduction of ignition pressures.

Engine T10050 included those design parameters deemed optimum from the fuel cooling parameter standpoint. The engine utilized 11% fuel cooling. Additional coolant (up to 25%) had demonstrated no additional chamber wall or chamber throat temperature reduction, a loss of 2 to 4 seconds in I_{sp}, and additional reduction in chamber flange temperature which it was felt was not warranted at the expense of performance. It also included an all steel oxidizer valve standoff. This configuration is shown in Figure 38.

FUEL COOLED ENGINE CONFIGURATIONS

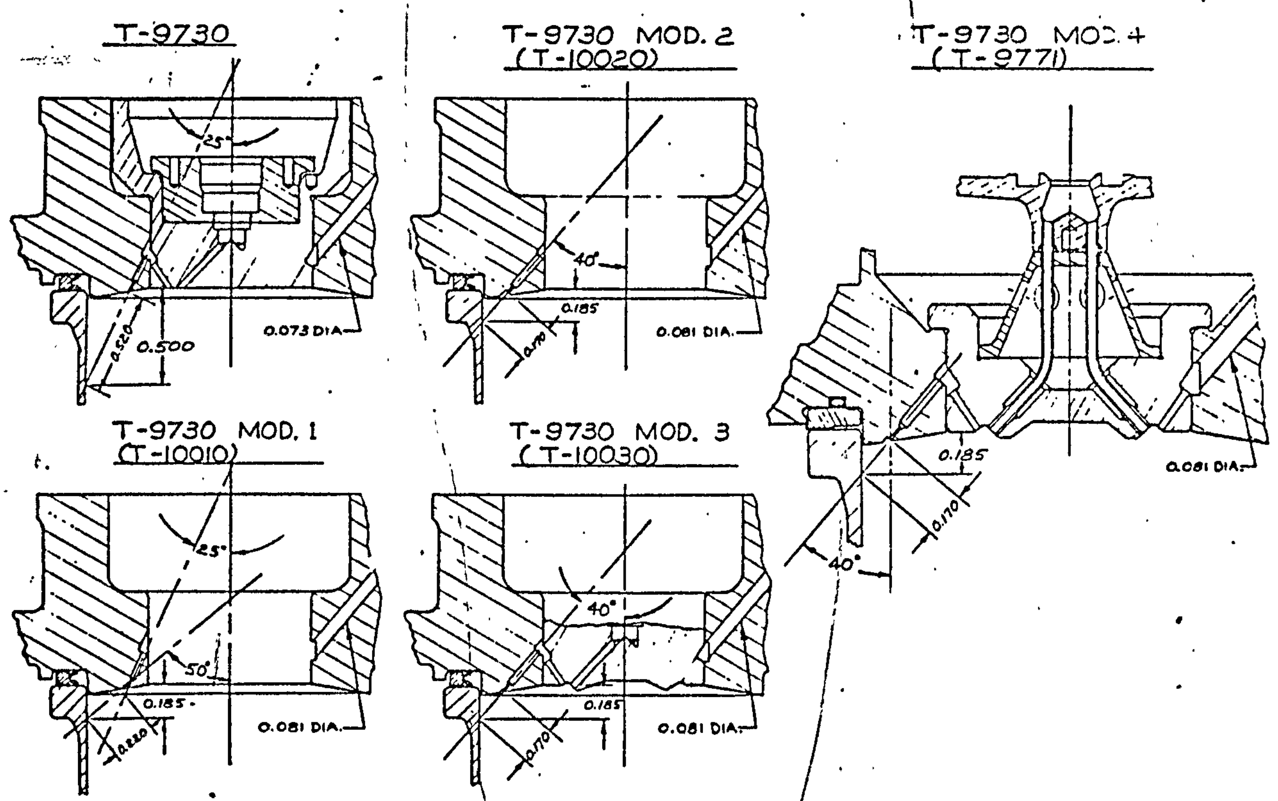
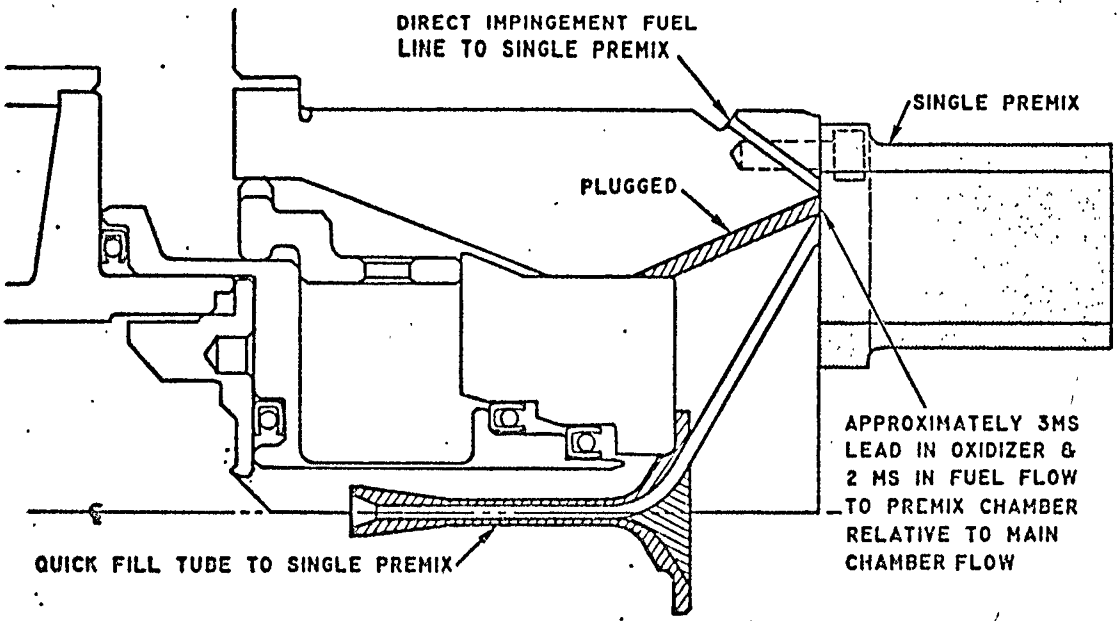


Figure 36

PRE-IGNITION CHAMBER T9640 CONFIGURATION



© FLAT FACE WITH EARLY FLOW PRE IGNITOR

Figure 37

T-10050 INJECTOR

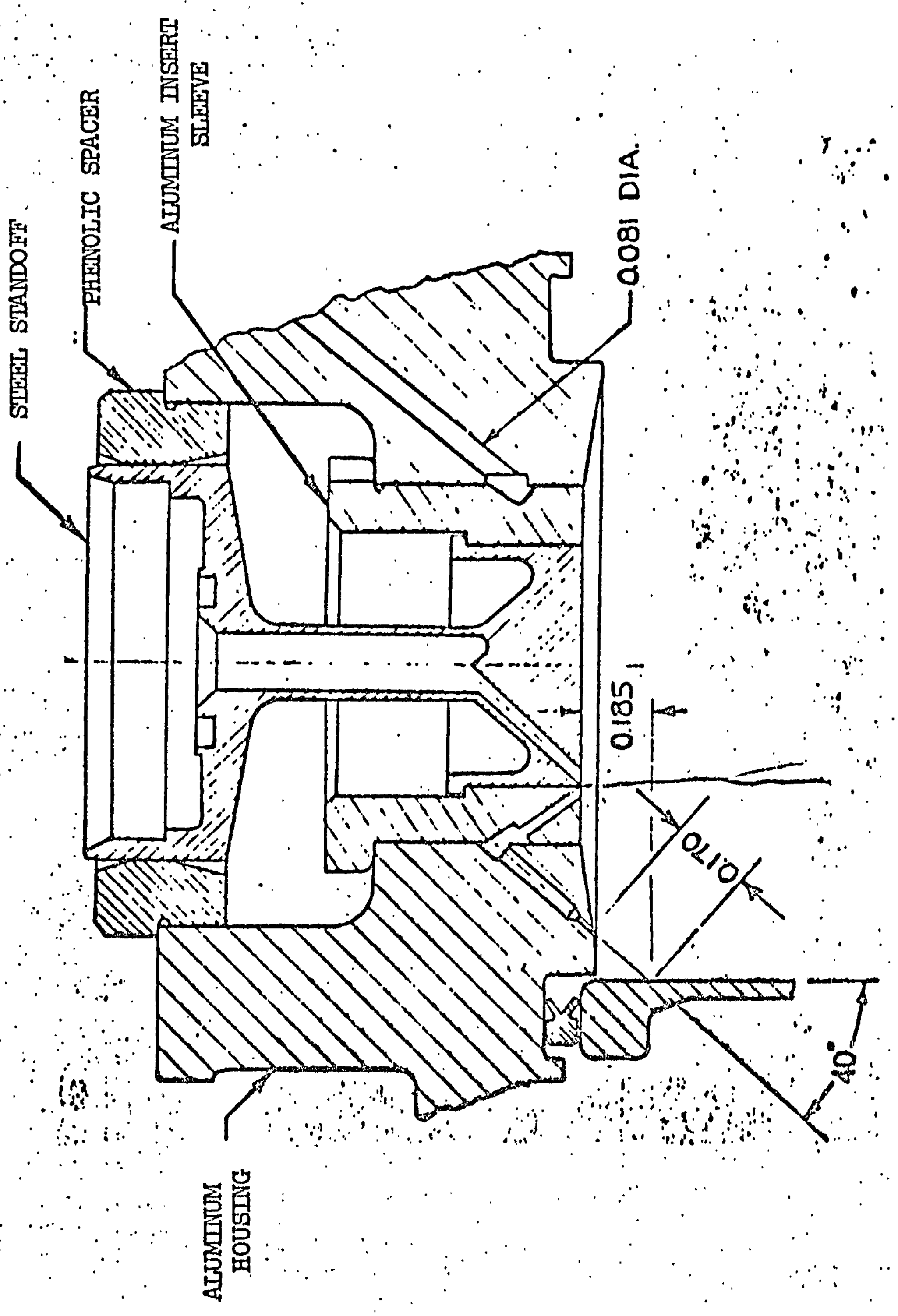


Figure 38

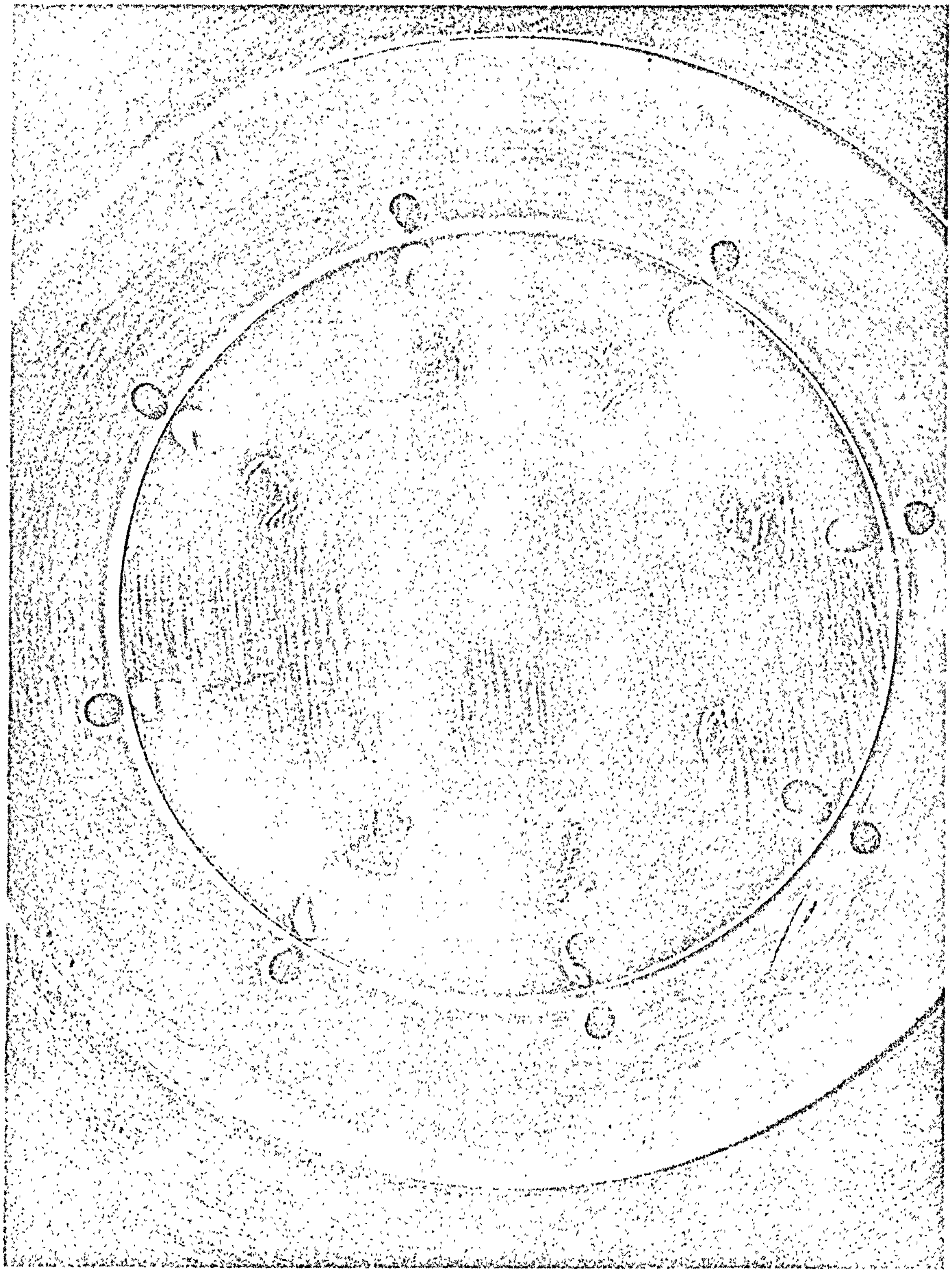
Figure 39 shows the condition of a T10050 engine injector face before testing. Engine T10050, S/N 0002 was tested and on 31 March 1964 the chamber shattered during a portion of the tests. At the time a total of 589 starts and 1993 seconds of burning, including one 90 second run, had been accumulated. Figures 40 and 41 show the injector face of T10050, S/N 0002 engine after the tests wherein the combustion chamber shattered. Note the distortion of the inner edge of the main fuel holes in the aluminum. Part of the aluminum at the fuel hole exits had been displaced in such a manner as to partially restrict the exits. It was believed that the injector hole deformation was the same type as previously experienced but was slightly different in formation due to the higher melting temperature of the steel enshrouding the oxidizer holes as compared to the aluminum around the fuel holes. It appeared to be due to small hydraulic reversed resultant that is formed, in addition to the forward resultant, when the two liquid streams are impinged together at a significant angle. In this case, the reversed resultant is made up of hypergolic liquids and hence high localized pressures and temperatures are created which can cause this type of damage. This damage causes "bushy" streams and resultant poor mixing.

Three factors may have contributed to the shattering of the chamber.

1. There was a small crack in the combustor chamber in the failure area, progressing from the outside surface inward.
2. The injector fuel holes had deteriorated to the point where good liquid-liquid mixing was not being obtained prior to the pulsing operation during which the failure occurred.
3. A facility propellant leak had cooled the combustor to approximately -65°F just prior to failure.

The bad fuel injection holes could have resulted in greater propellant accumulation and hence higher ignition peak pressures. The low combustion chamber temperature (unplanned) could have resulted in the accumulation of frozen propellant on the combustor prior to ignition, and a crack could have provided a weak point from which the failure could progress.

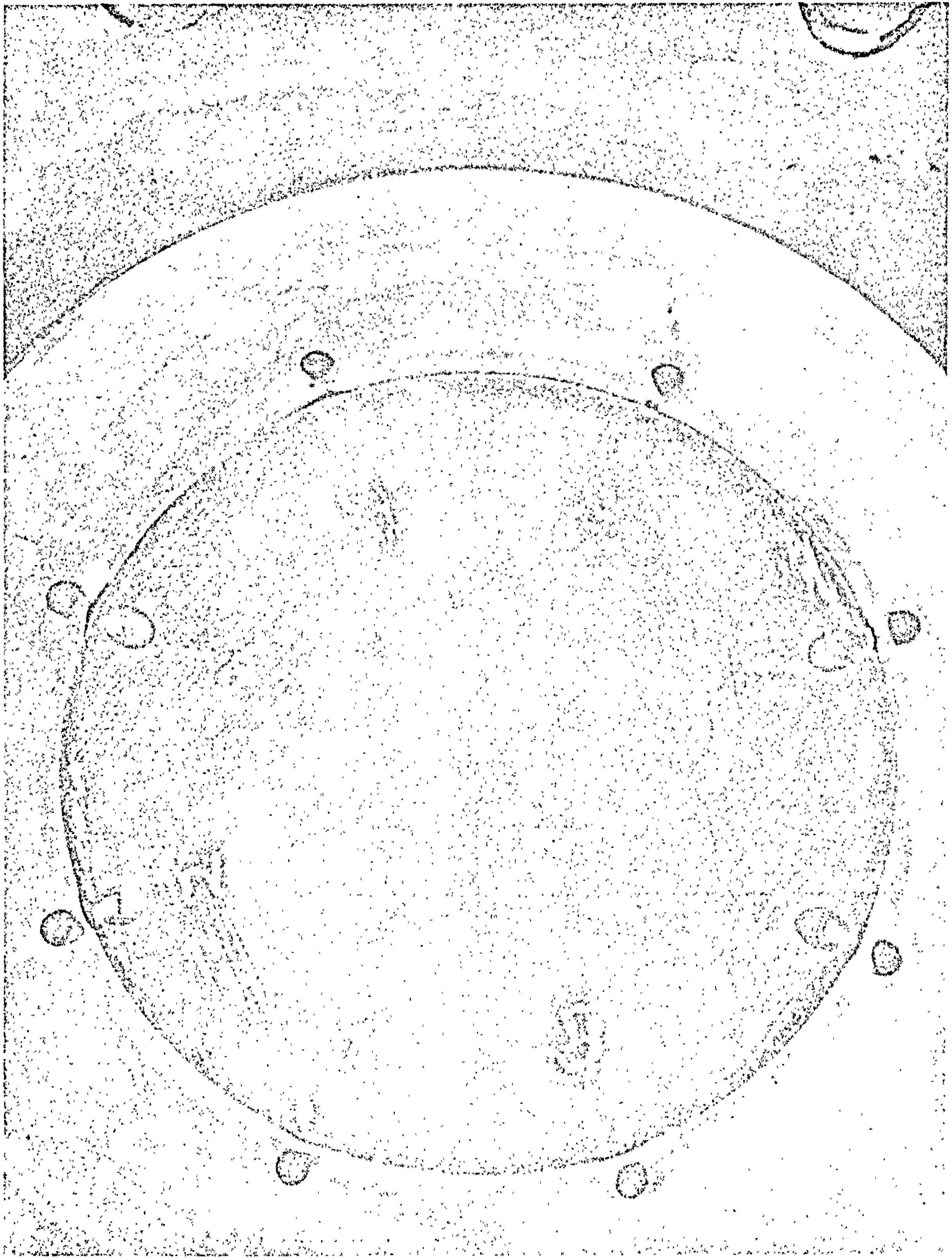
As a result of the damaged fuel holes, future engines incorporated a "V" groove between the fuel and oxidizer hole exits and subsequent tests showed no evidence of injector hole deterioration. Figure 42 shows the "V" groove configuration, T10060.



NEG. 6092-1

T-10050 Engine, S/N 0003 Injector Prior to Engine Altitude Testing

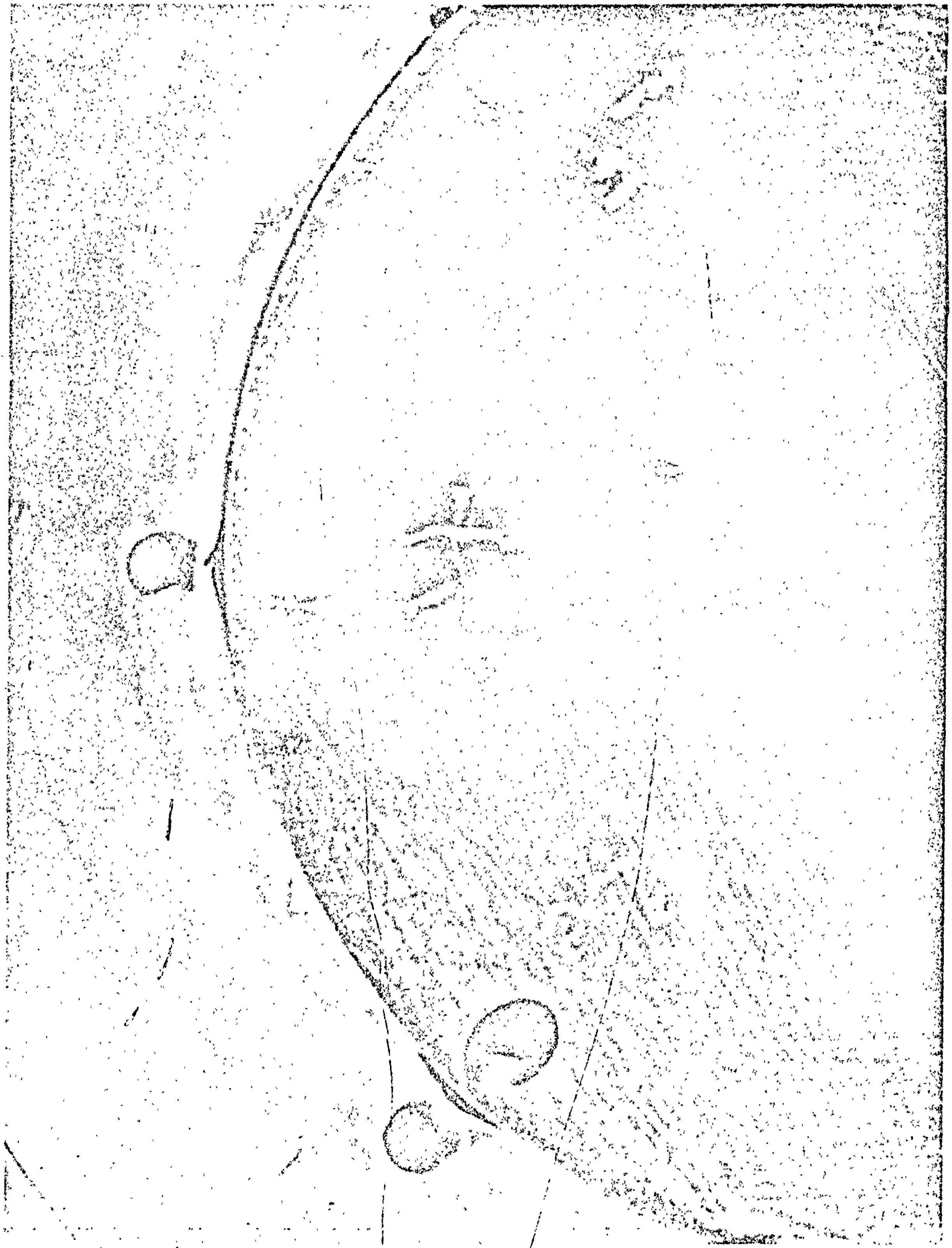
Figure 39



NEG. T3106-151

Engine T-10050, S/N 0002 Injector Face after Chamber Shatter, Fuel Hole Exists Distorted

Figure 40



NEG. CT3106-152

T-10050 Engine, S/N 0003 Injector Doublets No. 3 and No. 4 after Chamber Shatter, Fuel Hole Exists Distorted

Figure 41

T10060 INJECTOR

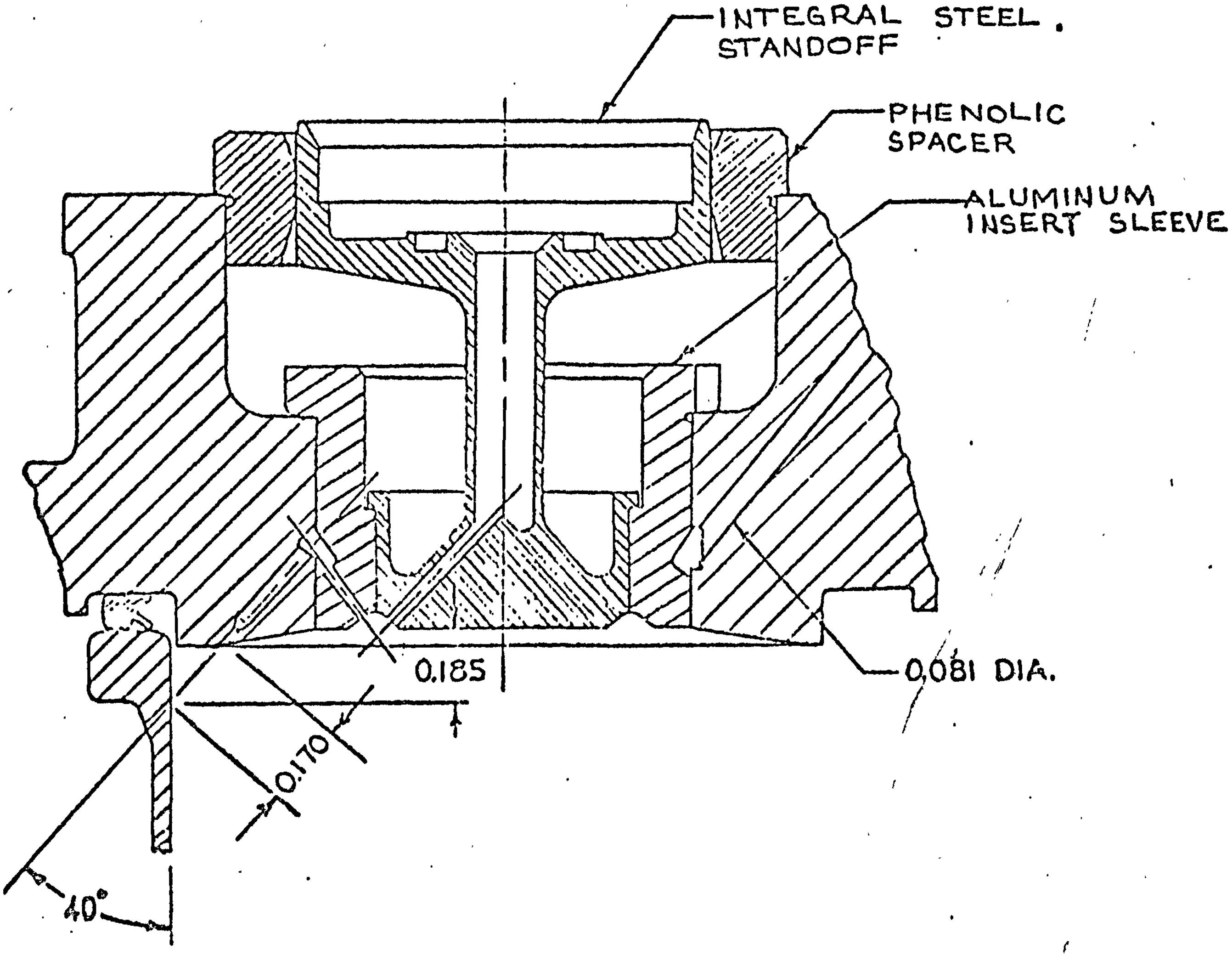


Figure 42

IV. PREIGNITER DEVELOPMENT

A NASA/North American/TMC meeting was held on 15 April 1964 to discuss the engine program as a result of the chamber shatter experienced on 31 March, 1964. As a result of this meeting, the following programs were undertaken:

1. Eliminate ignition spiking completely.
    - (a) Detonation Studies, Structure Dynamics, Understanding of Phenomenon.
    - (b) Fuel blends and additives.
    - (c) Preignition chamber.
    - (d) Off O/F ignition.
    - (e) Improved valve matching - Electrical, Mechanical.
2. Study methods to minimize ignition spikes.
    - (a) Chamber L*
    - (b) Chamber shapes.
    - (c) Fuel change.
    - (d) Injector Modification.
    - (e) Repeatability Documentation and separate variables.
    - (f) Thermal Management Analysis.
3. Determine possible alternate materials for combustion chamber fabrication.
    - (a) Pure moly
    - (b) TZM
    - (c) Titanium
    - (d) Columbium
    - (e) TD Nickel
    - (f) L605
4. Investigate performance improvement methods.
    - (a) Critical tests to verify design criteria.
    - (b) Conduct prototype testing.

A. Development

Testing effort was centered around the development of a preignitor. Both analysis and testing had shown that explosions of liquid and solid phase detonable products would not occur if the chamber pressure was above the vapor pressure of propellants prior to propellant injection. The preignitor produces a start-up pressure in the main chamber thus minimizing and/or eliminating ignition overpressure spikes.

Figure 43 shows a four valve preigniter test rig used to develop the preigniter concept and to obtain design criteria. Figures 44 and 45 present two of the early preigniter engine configurations tested; the preigniter used to evaluate heat transfer effects and the first preigniter injector.

In addition, testing had shown that ignition pressure spikes with fuel lead injection had much greater consistency and repeatability and were approximately one-half of the peak values recorded with oxidizer lead injection. This criteria lead to the valve coil design concept of fuel leads. Wherein the automatic coils of the fuel coil have fewer turns that the oxidizer coil and the coils are wired in parallel. On the direct coil, the fuel coil has more turns than the oxidizer and the coils are in series.

Off O/F tests were conducted and showed that ignition pressures were much lower for O/F's less than 0.5 and greater than 6.0, than the pressures experienced in the O/F range of 0.5 and 6.0.

During this period, analytical evaluations were made with respect to the rate of condensation of oxidizer on cold combustion wall, comparisons of heat sink properties of various thicknesses of cold combustion chamber walls, detonation analyses, literature searches, and consultations with authorities in the field of detonation. Experimental effort included development of a thin film platinum resistance thermometer, time reference instrumentation and streak photography. Testing included engine firings as well as cold flowing of propellants, and drop tests were made to determine the effect of impacting the propellants with one a liquid and the other a solid. Also obtained were indications of the explosive characteristics of 50/50, MMH, UDMH, and N_2H_4. Among the results noted--when liquid hydrazine is dropped onto solid N_2O_4, explosions occurred in 40% of the runs. The reverse situation caused no explosions. Solid mixture of N_2H_4 and N_2O_4 were made at approximately -60°F as a measure of chemical affinity of both propellants. Both propellants in cold solid form burned on contact.

A summary of the tests is tabulated below:

FOUR VALVE PRE-IGNITER TEST RIG

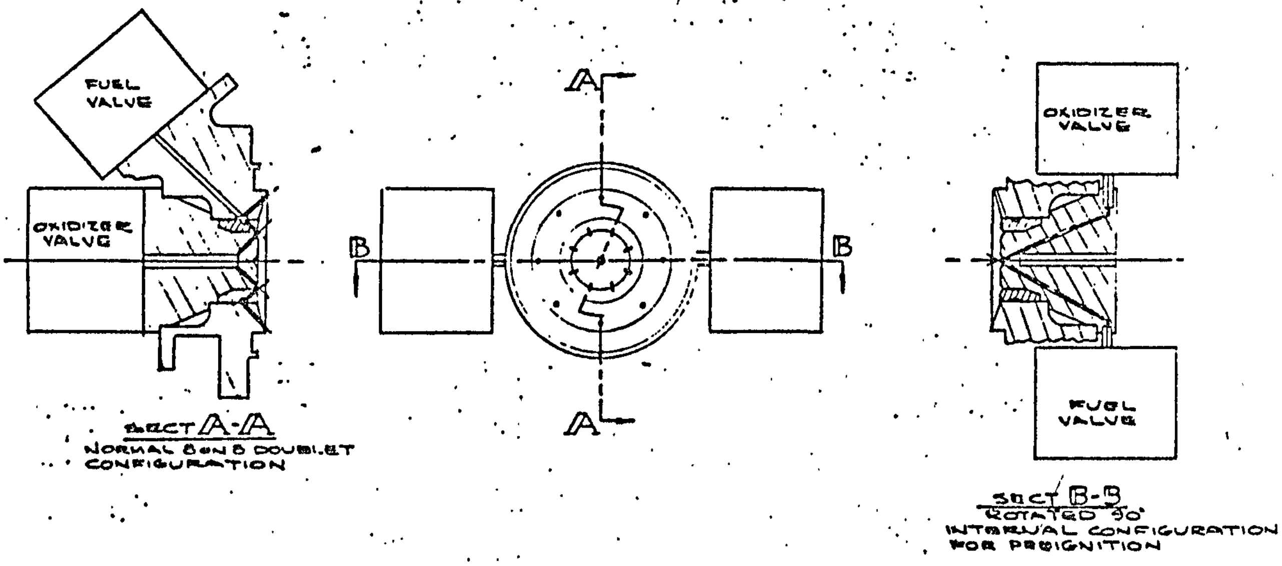


Figure 43

PRE-IGNITOR HEAT TRANSFER TEST

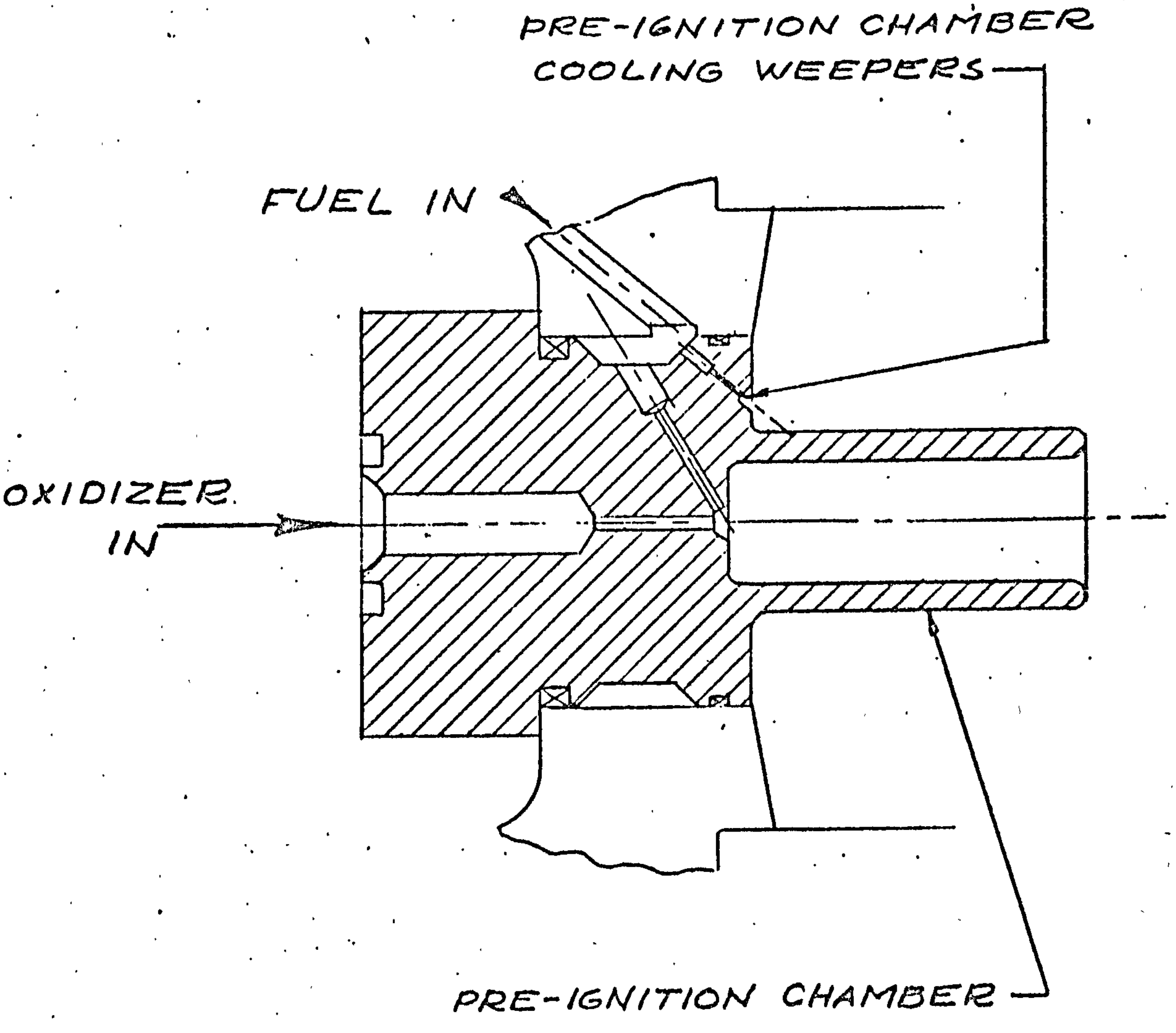


Figure 44

FIRST PRE-IGNITOR INJECTOR

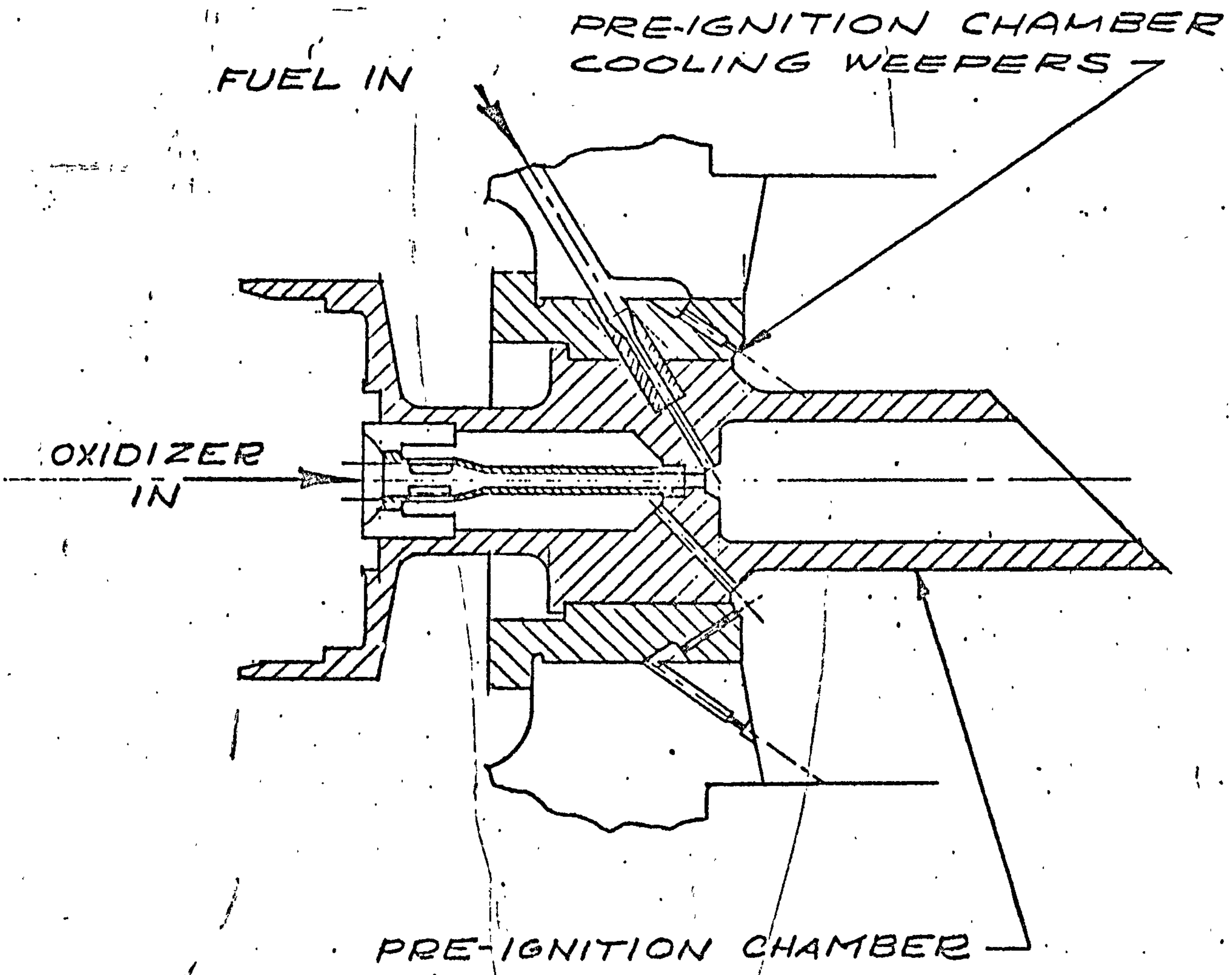


Figure 45

SUMMARY OF PROPELLANT DROP TESTS

	MMH		UDMH		N_2H_4		50/50	
	No. of Tests	No. of Exp's	No. of Tests	No. of Exp's	No. of Tests	No. of Exp's	No. of Tests	No. of Exp's
Dropping Liquid N_2O_4 onto Frozen Fuels	2	0	2	0	2	0	2	0
Dropping Liquid Fuels onto Frozen Oxidizer	2	0	2	0	5	2	2	0

NOTE:

1. All drops made from a height of 8 inches.
2. All combinations produced much heat due to a vigorous chemical reaction.
3. Mixing frozen N_2H_4 particles with frozen N_2O_4 particles at approximately -60°F produced immediate burning. A flame was noted but no explosion occurred.

Fuel additives were ordered and received, however, they were not tested on the program but were subsequently tested on another NASA program. The additives were each mixed with Aerozine-50 (one percent by weight of additive) and supplied by the Callery Chemical Company. The additives were:

N-Allylaneline
Pyrolle
Phenylsulfide
Phenylhydrazine
Cyclopentadienyliron
Methylbutynol
P-Xlidine
4-%-Butylthiophenol
Allylphenylether
Furfural alcohol
Oxamide Dihydrazone
Melamine

None of these had a significant effect on ignition overpressures.

Injector testing centered around the 8 on 8 fuel cooled engine configuration, however, some testing was conducted on a T9930 engine which had a triplet injector (16 on 8). The schematic and the injector face are shown in Figures 46 and 47, respectively. It was expected that the engine would demonstrate increased steady state I_{sp} due to better mixing and to have a more constant performance level as a function of O/F ratio because of the constant momemtum angle furnished by the triplet mode of injection. Test results indicated the engine ran at two performance levels, one at low performance with I_{sp} = 273 seconds at an O/F of 2.0 and a higher level with burning conditions resembling high heat transfer burning with wall temperatures too high to continue testing. I_{sp} was 313 seconds at an O/F of 2.0. Further testing was discontinued on this configuration due to the high temperatures encountered.

Under the alternate materials program considerable laboratory testing such as torch tests were conducted as well as actual engine tests on chamber materials of 90 tantalum-10 tungsten coated with both disilicide and tin-aluminide, see Figures 48 and 49, respectively. (These were chambers fabricated under the early Apollo Program - Alternate Materials Program), two piece TZM combustor/L605 bell and two piece moly combustor/L605 bell. During this time period, Summer 1964, the design criteria changed from a one piece ribbed combustor chamber to a two piece combustor-bell assembly. The combustion chamber could be fabricated of various materials while the bell material selected was L605. This two piece design was the result of:

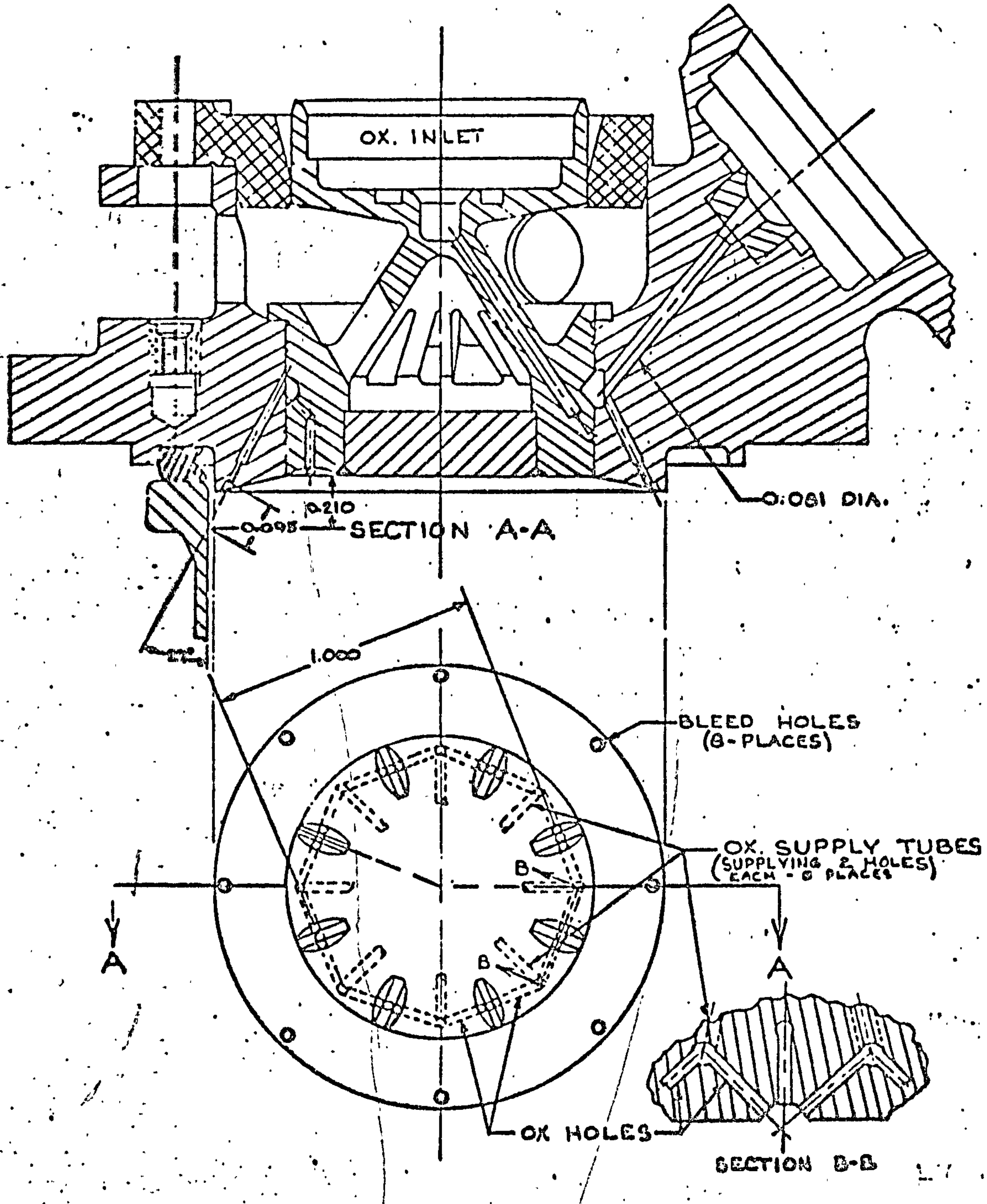
1. Several bells on the one piece molybdenum had been cracked due mainly to handling and installation difficulties. The bell fabricated of L605 was much stronger and had far greater ductility.

2. The L605 bell was no more difficult to fabricate than the moly bell.

3. The L605 bell could be reinforced for boost air loads by the use of machined ribs on the external surface and the strength to weight ratio was much better than a corresponding moly bell.

Results of the tests of the alternate materials was encouraging. However, because of the immense experience available with molybdenum, it was selected as the qualification material.

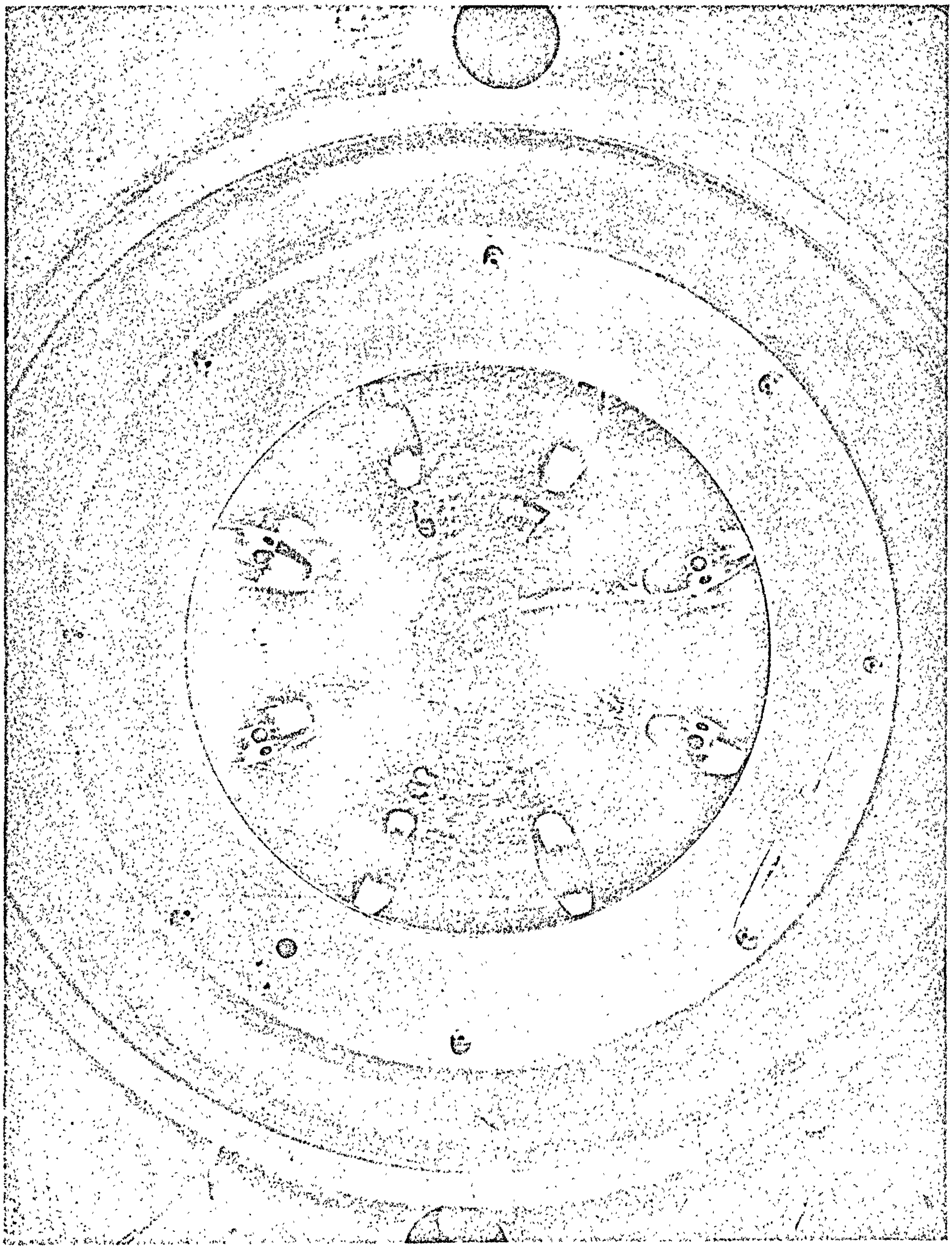
During July 1964, it was agreed that all injector effort would center around a preigniter engine with fuel cooling, ribbed two piece moly combustor/L605 bell. This basic configuration is shown in Figure 50.



CONSTANT MOMENTUM ANGLE INJECTOR ASS'Y

T9930

Figure 46



NEG. T3106-167

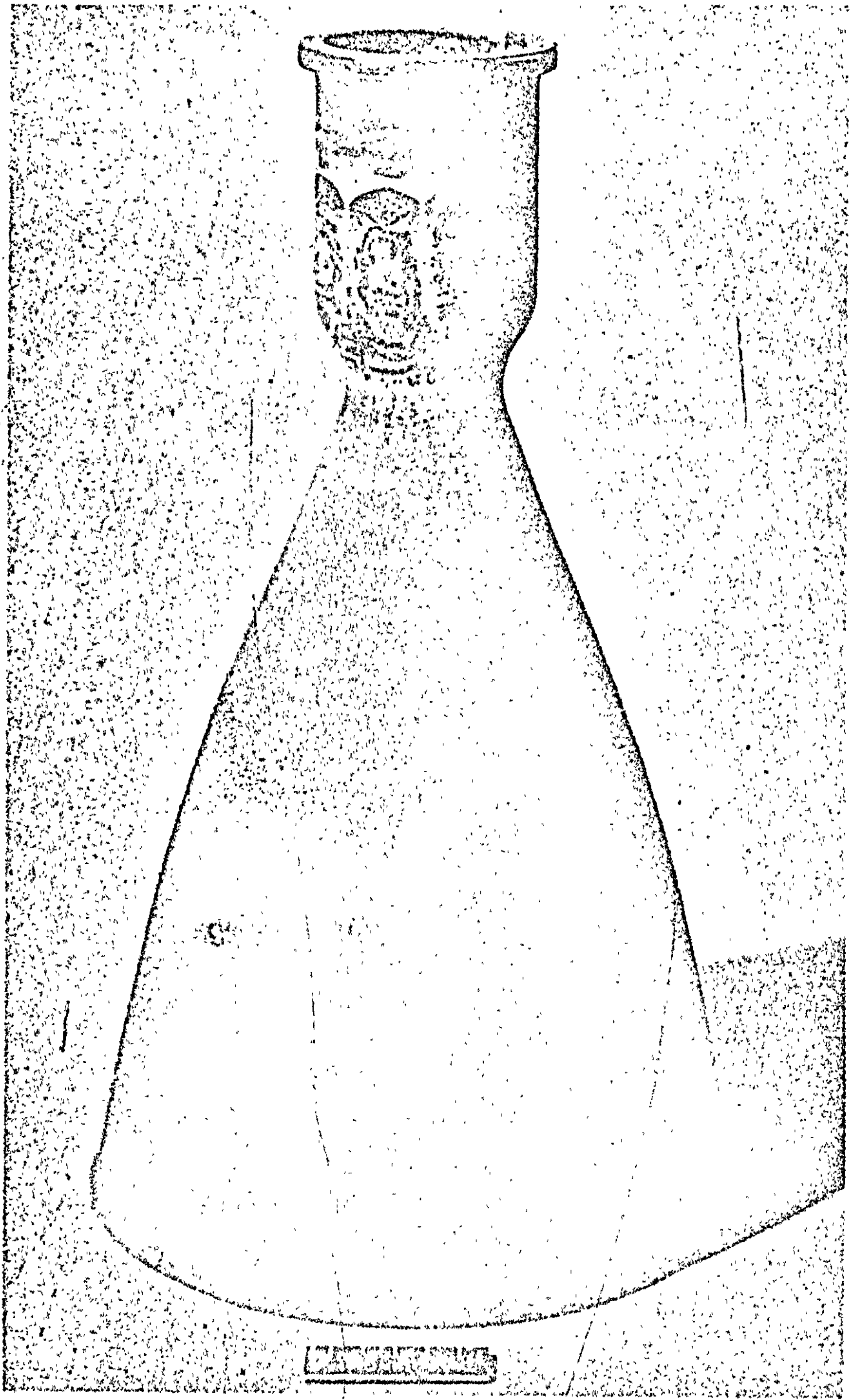
8 Triplet Injector Head Assembly Face - Preburn

Figure 47

NEG. 6162-1

MD 1141, S/N 001 Disilicide Coated 90 Ta-10W Altitude Combustion Chamber

Figure 48



NEG. C9656-1

MD1141, S/N 002 Tin-Aluminide Coated Altitude Combustion Chamber from T-10060 Engine, S/N 0004 - Total Burn Time 7972 Seconds, Overall View Showing Coating Condition

PREIGNITER ENGINE T10650

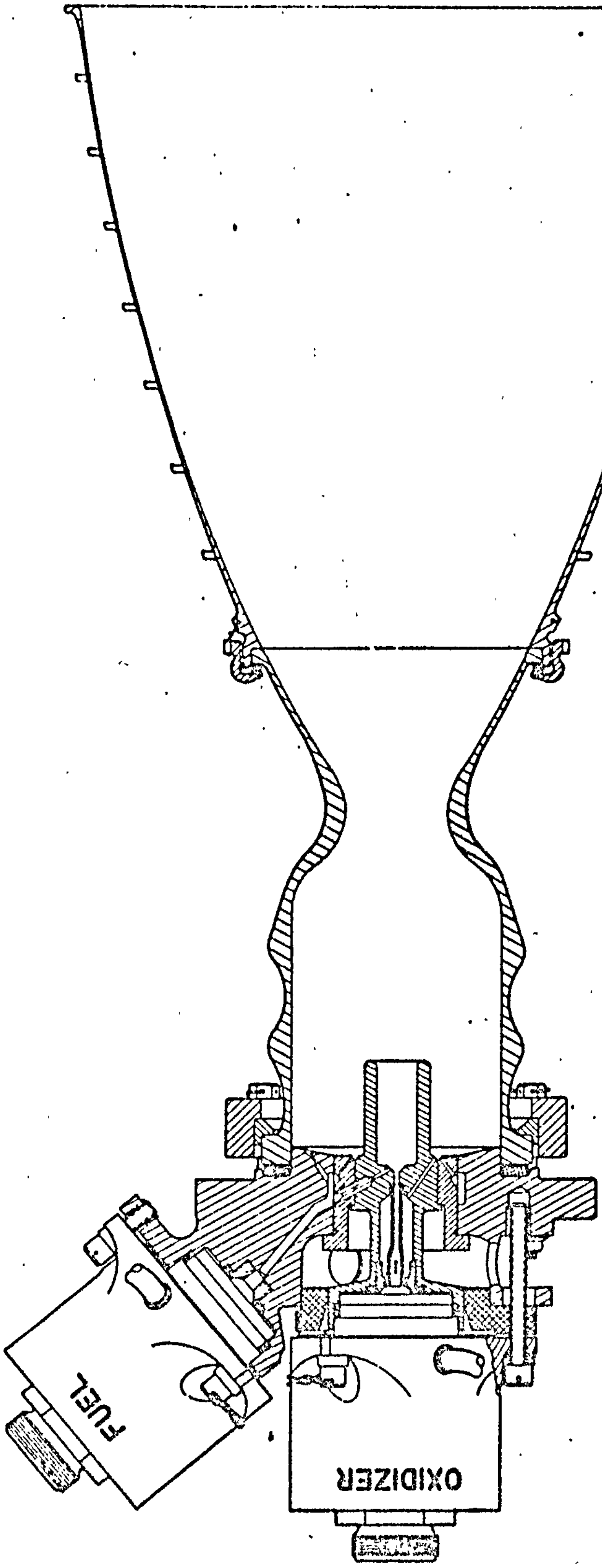


Figure 50

The development effort centered around optimizing preigniter cooling. The T10650 engine configuration was tested for both ignition tests as well as steady state tests. The engine satisfactorily demonstrated a continuous run of 500 seconds thus meeting this specification requirement.

Additional testing was conducted to determine the preigniter tubes flow "window" area to obtain the optimum preigniter flow rates and O/F for repeatable preigniter ignition and stable, non-chugging, preigniter operation. A computer program was formulated and run to support the test program on optimizing the preigniter design.

Spikes with the early preigniter configuration occurred when:

1. Preigniter failed to light.
2. Spike on preigniter ignition.
3. Main chamber spike after preigniter ignition due probably to inadequate pressurization and residual propellants.
4. A spike after normal preigniter ignition but before main chamber propellants enter the chamber.

Parameters such as preigniter O/F, flow, and injection pressure were varied to optimize preigniter performance. Ignition limits of the preigniter could be extended by lowering the preigniter O/F. Combustion stability (lack of chugging) of the preigniter is governed by the ratio of the junction pressure to the preigniter combustion chamber pressure and requires that the junction pressure be as high as possible within the limits of the engine supply pressure.

B. PFRT Configuration

During September 1964, the Pre-Qualification and/or PFRT configuration was selected. This configuration is shown in Figure 51.

The general characteristics of the engine were as follows:

1. I_{sp} (steady state) = 286 ± 6 seconds.
2. I_{sp} (minimum impulse bit) = 130 seconds.
3. No severe (>400 psi) start transients for propellant leads 2 ms Oxidizer and 20 ms Fuel.
4. No engine temperatures in excess of design values.

PFRT CONFIGURATION

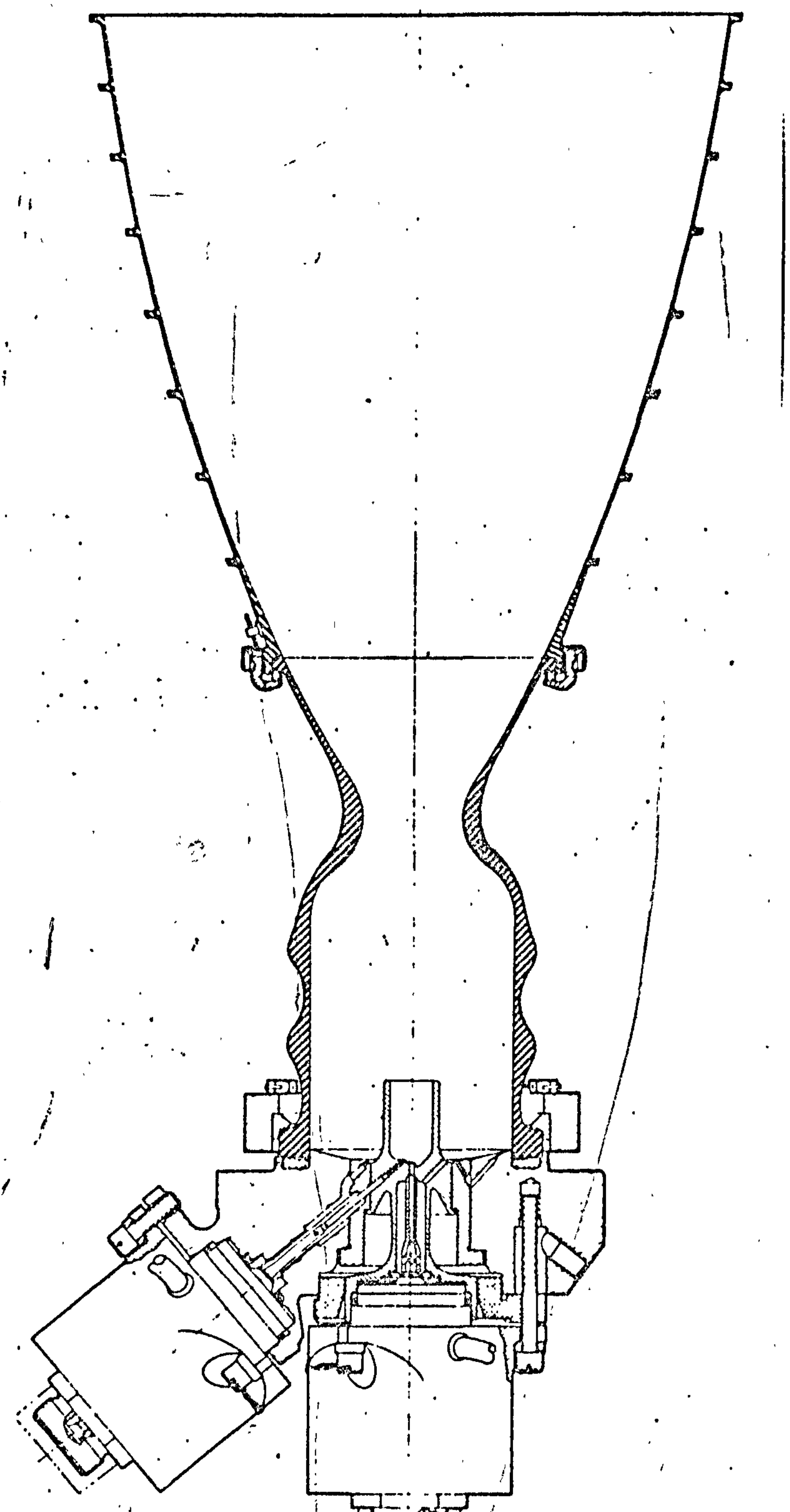


Figure 51

During preigniter testing for ignition transients in September 1964, it became quite evident that the first few pulses of an engine run wherein the propellant lines had recently been primed, resulted in improper preigniter operation. Analysis of the causes for this resulted in evidence that gas bubbles form in the propellant lines during engine priming and these bubbles interfere with proper preigniter operation during these first few pulses. In addition, while running hot propellants (100°F+) it was noted that oxidizer flows would drop off on each succeeding run by a significant amount (0.005 pps). After about 30 to 45 minutes under pressurization at about 170 psig, the flow would stabilize.

Investigation revealed that nitrogen solution or entrainment was the problem. In order to partially eliminate the problem, a float was installed in the oxidizer pulse tanks in the cells to limit the area of hot propellant exposed to pressure. The float appeared to be effective in the subsequent tests since the flow of hot oxidizer was now repeatable from run to run.

Degassing of propellants also was accomplished at this time by heating the propellants to about 100°F under low (very near atmospheric) pressure. Once degassed, proper preigniter operation was always achieved.

A preliminary engine study program was completed covering Passive Thermal Control (PTC) involving valve heat loss and injector head heat loss and submitted to North American during September 1964. This program had been under contract since 18 June 1964. TMC Report S-444 dated September 12, 1964 presented the results of this study.

During October 1964, the program was oriented into a two phase program:

1. A three engine Preliminary Flight Rating Test (PFRT) to cover non-passive thermal control engines to be delivered for flight (in actuality on AF 009).
2. A program covering development and Qualification of a Passive Thermal Control (PTC) Engine.

PFRT configuration testing continued demonstrating sufficient temperature margins in all areas. Some of the temperatures were as follows:

Parameter	Steady State Operating Temperature °F	Maximum Allowable Temperature °F
Chamber Throat	2200	2900
Bell Attach Nut	1550	1750
Bell Nozzle Exit	1625	2200
Bell at Joint	1850	1880
Injector Head Soakback	300	350

Figures 52, 53 and 54 present typical performance curves of this engine configuration.

Acceptance tests on the first PFRT engine P/N X21424, S/N 0001 were conducted on October 21, 1964, and S/N 0002 was acceptance tested on November 14, 1964. One of the PFRT engines, S/N 0002, was subjected to electrical bonding tests (not as a part of the PFRT program but rather as a supplemental test.)

Resistances for all mechanical discontinuities were measured. The combustor offered no convenient contact surface for the electrical clip leads used and contact resistance was relatively high, however, the engine was within specification limitations. Figure 55 presents the results of the test.

The Performance Flight Rating Test (PFRT) to Marquardt Test Plan (MTP) 0019P was a three engine test. Its objective was to demonstrate Engine P/N 227486 performance level and suitability for unmanned flight test within the nominal performance requirements of NR Procurement Specification MC 901-0004D. The three 227486 engines selected for this test were instrumented after acceptance test and reidentified as P/N X21424, S/N's 0001, 0002, and 0003. The PFRT Program consisted of essentially two types of tests; environmental or engine structural tests and engine firing tests. These tests were followed by engine disassembly and inspection. Figure 56 shows the tests and order of testing conducted in each of the three engines. A short description of each test is included below.

1. Environmental Tests

Environmental tests conducted to establish engine structural integrity included the following:

(a) Boost Vibration Tests - Appendix G of MTP 0019P
To demonstrate structural integrity of the engine when subjected to the maximum vibration loads expected during the Apollo launch.

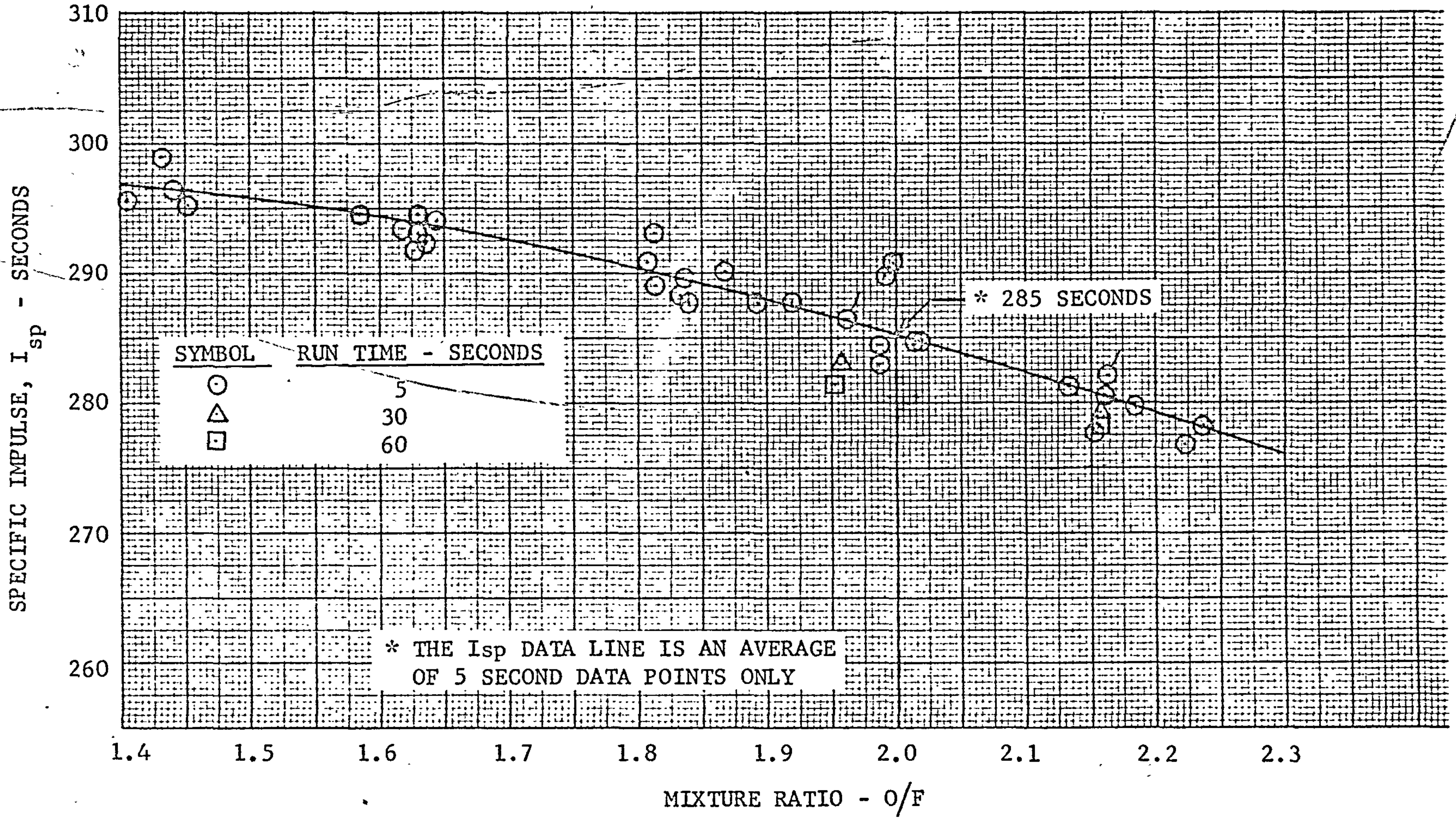
(b) Salt Fog Test - Appendix H of MTP 0019P
To demonstrate engine contamination and corrosion resistance to a salt fog atmosphere.

(c) Water Flow and Valve Performance - Appendix K of MTP 0019P
To provide checks of valve response, valve electrical characteristics and integrity of valve and other engine seals at one or more times during the test program.

(d) Static Load (Limit) - Appendix T of MTP 0019P
To demonstrate engine structural integrity under maximum expected boost phase airloads.

SPECIFIC IMPULSE VS MIXTURE RATIO

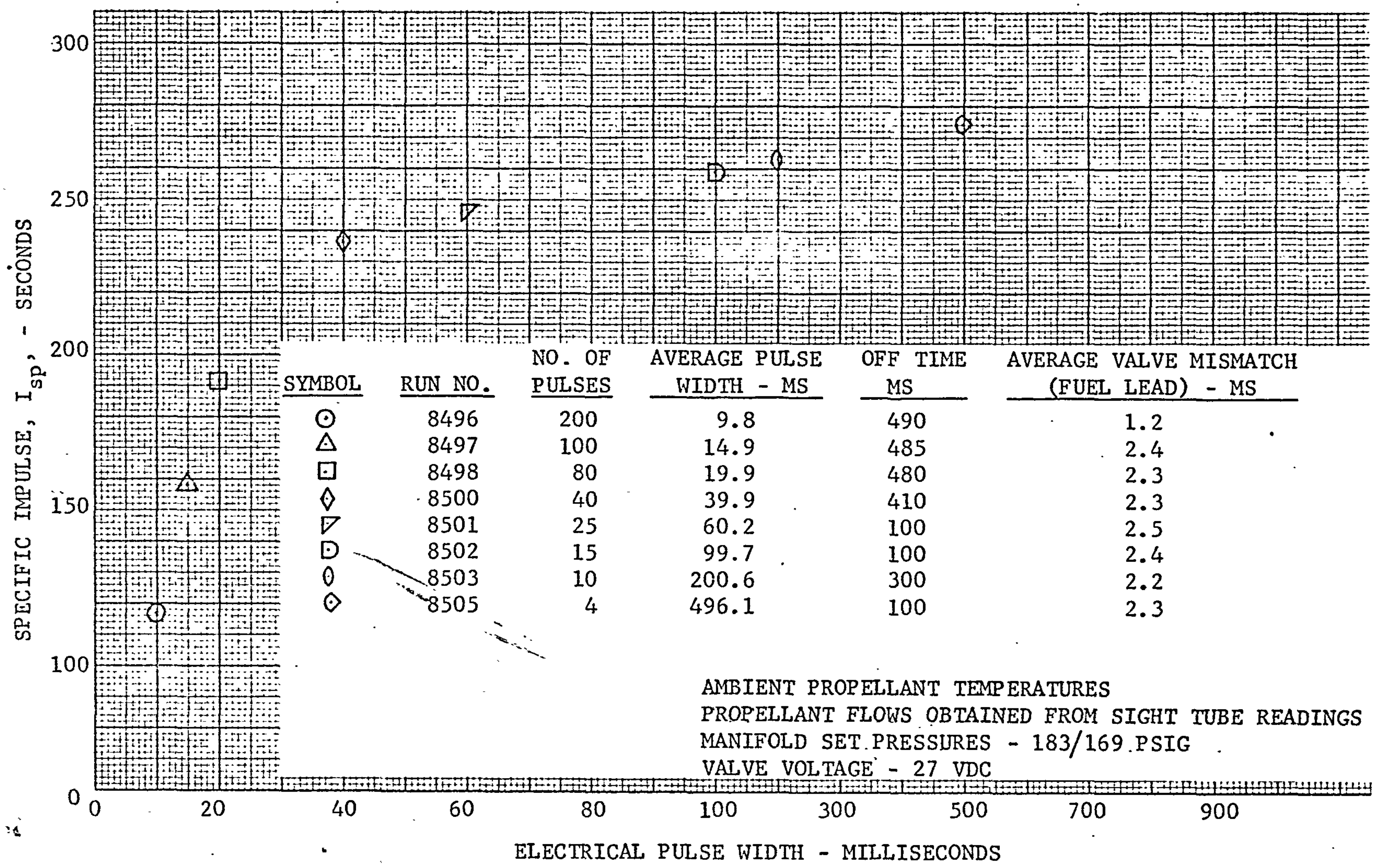
PFRT CONFIGURATION, PRE-IGNITER ENGINE RUNS 8409-8444
TEST NO. 3201, CELL 1, RUN DATE: 9/24/64
ENGINE ASSEMBLY: T-10670, S/N 0002-3
CHAMBER P/N T-10676/T-10188, S/N 002/004
2 PIECE RIBBED MOLY CHAMBER /L-605 SKIRT
24.6% FUEL BLEED FLOW



* THE Isp DATA LINE IS AN AVERAGE OF 5 SECOND DATA POINTS ONLY

Figure 52

PULSE SPECIFIC IMPULSE VS ELECTRICAL PULSE WIDTH
PFRT CONFIGURATION, CALIBRATION TEST RUNS
TEST NUMBER: 3201, CELL 1, RUN DATE: 9/29/64
ENGINE P/N T-10670, S/N 0002-3
CHAMBER P/N T-10176/T-10188, S/N 002/004
2 PIECE RIBBED MOLY CHAMBER / L-605 SKIRT
24.6% FUEL BLEED FLOW



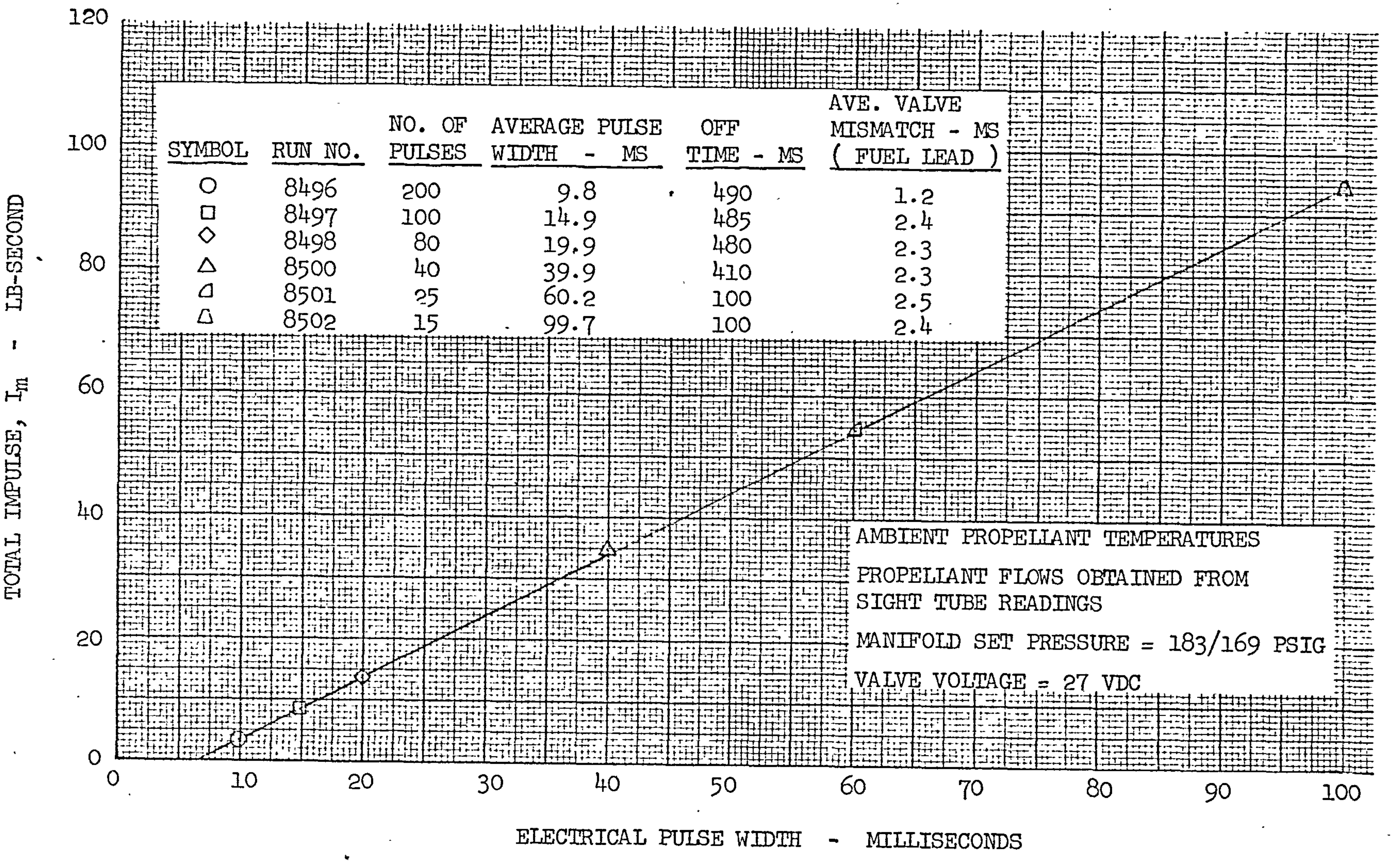
SYMBOL	RUN NO.	NO. OF PULSES	AVERAGE PULSE WIDTH - MS	OFF TIME MS	AVERAGE VALVE MISMATCH (FUEL LEAD) - MS
⊙	8496	200	9.8	490	1.2
△	8497	100	14.9	485	2.4
⊡	8498	80	19.9	480	2.3
◊	8500	40	39.9	410	2.3
▽	8501	25	60.2	100	2.5
D	8502	15	99.7	100	2.4
0	8503	10	200.6	300	2.2
◈	8505	4	496.1	100	2.3

AMBIENT PROPELLANT TEMPERATURES
PROPELLANT FLOWS OBTAINED FROM SIGHT TUBE READINGS
MANIFOLD SET PRESSURES - 183/169 PSIG
VALVE VOLTAGE - 27 VDC

Figure 53

PULSE TOTAL IMPULSE vs. ELECTRICAL PULSE WIDTH

PFRT CONFIGURATION, CALIBRATION TEST RUNS
TEST NO. 3201, CELL NO. 1, RUN DATE: 9-29-64
ENGINE P/N T-10670, S/N 0002-3
CHAMBER P/N T-10176/T-10188, S/N 002/ 004
2 PIECE RIBBED MOLY CHAMBER/ L-605 SKIRT
24.6% FUEL BLEED FLOW



SYMBOL	RUN NO.	NO. OF PULSES	AVERAGE PULSE WIDTH - MS	OFF TIME - MS	AVE. VALVE MISMATCH - MS (FUEL LEAD)
○	8496	200	9.8	490	1.2
□	8497	100	14.9	485	2.4
◇	8498	80	19.9	480	2.3
△	8500	40	39.9	410	2.3
◁	8501	25	60.2	100	2.5
△	8502	15	99.7	100	2.4

AMBIENT PROPELLANT TEMPERATURES

PROPELLANT FLOWS OBTAINED FROM SIGHT TUBE READINGS

MANIFOLD SET PRESSURE = 183/169 PSIG

VALVE VOLTAGE = 27 VDC

Figure 54

P/N X21424, S/N 0002 ELECTRICAL BONDING

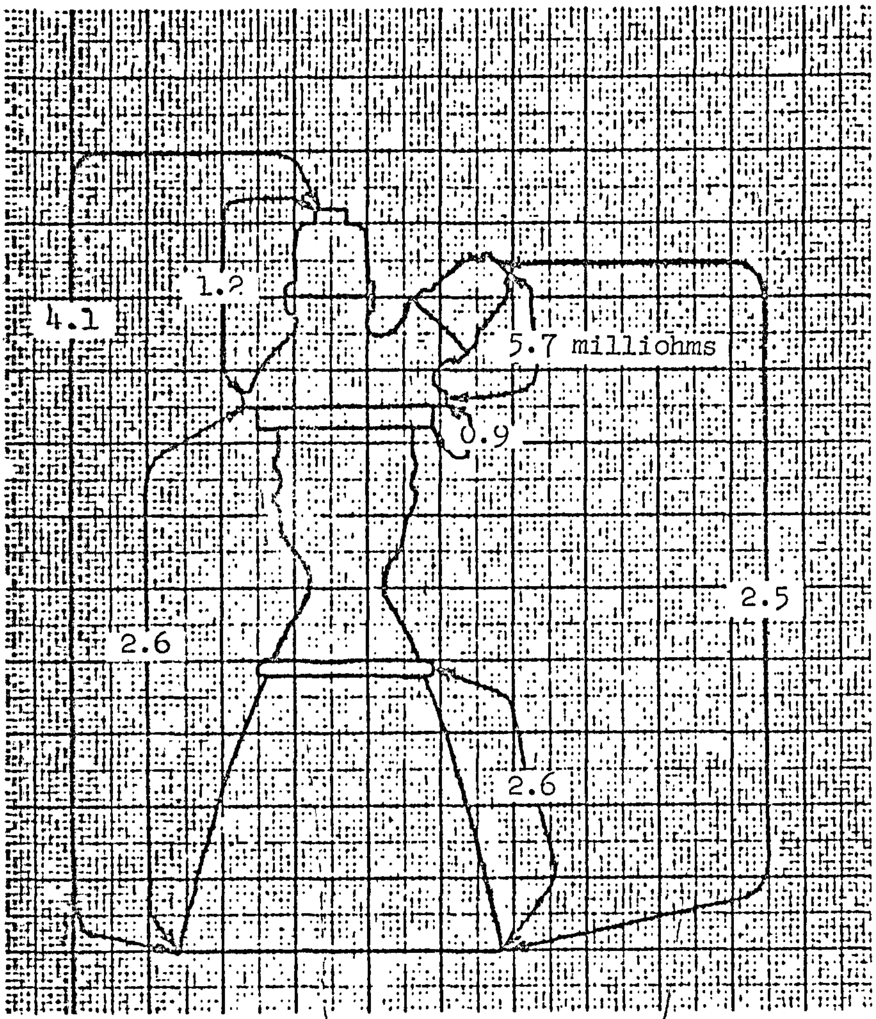


Figure 55

PFRT TEST PROGRAM

Test	Eng. No. I	Eng. No. II	Eng. No. III
Calibration Test Appendix A	1, 5	7, 11	7, 11
Pulse Operation Survey Test Appendix B	2	10	8
Boost Vibration Test Appendix G		1, 5	3
Salt Fog Test Appendix H			1
Water Flow and Valve Performance Appendix K	6	2, 4*, 6, 12	2, 4, 6, 12
Direct Coil Duty Cycle Test Appendix N	3	8	9
Static Load to Destruction Test Appendix S		13	
Static Load Limit Test Appendix T		3	5
Teardown Evaluation Appendix V	7	14	13
Special Duty Cycle Test Appendix W	4	9	10

*Valve Rebuild following this test.

Appendices noted are to Marquardt Test Plan (MTP) 0019.

Figure 56

(e) Static Load (Destruct) - Appendix S of MTP 0019P
To demonstrate engine structural margin beyond the maximum specification airload requirements.

2. Engine Firing Tests

Engine firing tests conducted to demonstrate engine operational capability included the following tests. All engines were tested to all of the firing tests.

(a) Calibration Tests - Appendix A of MTP 0019P
Calibration tests were run prior to and following all other firing tests. These tests documented engine performance at pulse widths from 13 milliseconds to 5.0 seconds.

(b) Pulse Operation Survey Tests - Appendix B of MTP 0019P
These tests demonstrated operational pulse capability. Pulse widths of 13 to 500 milliseconds were run. Repetition rates resulted in engine off times from 8 to 300 milliseconds.

(c) Direct Coil Duty Cycle Tests - Appendix N of MTP 0019P
These tests demonstrated engine operational capability using the valve direct coils. The test included operation under pressure transients from relief to nominal operating pressures. Further demonstration of "off-design" capability was shown by commanding engine starts at the voltage extremes (21 to 32 volts dc) under cold environmental conditions.

(d) Special Duty Cycle Tests - Appendix W of MTP 0019P
The Special Duty Cycle Tests demonstrated engine capability to perform specific duties for limited application usage. The test required performance of specific maneuvers consisting of commands varying from 13 milliseconds to 60.0 seconds, including firings from a temperature soakback condition.

3. Engine Teardown

Following completion of all testing, each engine was disassembled to the lowest practical component level. Detailed visual examinations were made of all components to determine possible structural degradation. Measurements were made of all critical areas, e.g., valve and chamber attach bolt torques, bell to combustor attach nut torque, etc. In addition, an assessment was made ty TMC Reliability of the remaining functional capability of the hardware.

PFRT testing on S/N 0001 engine was essentially completed in January 1965 (Testing initiated on 20 November 1964).

Engine S/N 0002 was removed from the PFRT test program in January 1965 in order to incorporate the latest solenoid valve pintle configuration described below. Valve leakage during various development tests had necessitated the need for an expanded valve development program to isolate the problem and eliminate the leakage. Early assessment of the problem indicated that galling of the armature pintle and the valve seat was the prime reason for leakage of the valves. Valve design changes made at this point to correct the leakage consisted of returning to the hardened armature pintle (previous history indicated there was no galling when hardened pintles were used) and incorporating specific visual inspection requirements for detecting burrs on valve seats. Evaluation tests were conducted on valves incorporating these changes and demonstrated that the changes were desirable.

In addition, a valve development program was initiated to evaluate other valve variables which could substantially influence valve leakage. Among these variables were properties of teflon under operational conditions and the dimensional variation in the teflon during a life cycle. Other variables evaluated included poppet geometry (cone versus spherical), valve dimensions (loose vs. tight) and impact eccentricities.

Rework of S/N 0002 valves incorporating the hardened poppet change were completed and the engine was acceptance tested on 28 March 1965 and it was reinstated into the PFRT program. Completion of PFRT testing was accomplished in late May 1965.

Engine S/N 0003, modified to include the hardened poppet changes, was acceptance tested on 20 March 1965 and the engine was placed into the PFRT Test Program. Testing was completed by mid-May, 1965. Final teardown, inspection and final evaluation of all three engines was accomplished by 25 June 1965.

Figure 57 presents the planned test plan requirements with respect to valve actuations and burn time. Figure 58 presents the actual operational summary and it may be noted that all planned requirements were exceeded.

All phases of the PFRT program were successfully completed. The engine demonstrated the capability to reliably meet the requirements for boost vibration, boost air loads and corrosive salt fog environment. Engine firing tests adequately demonstrated the required operational levels as well as the required structural levels under firing conditions. The engines, under periodic checks, demonstrated a high level of consistency in operational characteristics with no degradation of any parts.

OVERALL PFRT PLAN REQUIREMENTS
VALVE ACTUATIONS AND BURN TIME

Test Appendix	No. Times Performed			Valve Actuations			Sec. Burn Time		
	Eng. I	Eng. II	Eng. III	Eng. I	Eng. II	Eng. III	Eng. I	Eng. II	Eng. III
A	2	2	2	1738	1738	1738	102	102	102
B	1	1	1	3600	3600	3600	462	462	462
G	0	1	1	0	20	20	0	0	0
H	0	0	1	0	0	0	0	0	0
K	1	2	3	25	50	100	0	0	0
N	1	1	1	31	31	31	20	20	20
S	0	1	0	0	0	0	0	0	0
T	0	1	1	0	0	0	0	0	0
V	1	1	1	0	0	0	0	0	0
W	1	1	1	49	49	49	109	109	109
Est. Trim Runs				394	394	394	24	24	24
Totals				5837	5882	5922	717	717	717

Figure 57

PFRT ENGINE OPERATIONAL SUMMARY
ACTUAL EXPERIENCE

	Engine No. I	Engine No. II	Engine No. III
Total Burn Time (sec.)	1297.6	823.7	1207.3
Total Number of Firings	8548	6111	9699
Total Propellant Exposure (hrs.)-(min.)	150 hr.-43 min.	102 hr.-25 min.	134 hr.-47 min.
Total Oxidizer Valve Actuations	8665	6284	9973
Total Fuel Valve Actuations	8648*	6397	10014

All values shown are exclusive of acceptance test.

*Total valve cycles (fuel to oxidizer for one engine) different due to purging, bench testing or valve checkouts.

Figure 58

Final teardown inspection of the engines revealed normal, expected wear characteristics with no indication of structural degradation which could lead to a performance or reliability reduction. All engines were capable of further operation, indicating operational capability in excess of that required. Figures 59 through 64 present some of the pertinent performance results obtained during the program on the three engines. Performance in every case met or exceeded specification and Test Plan requirements. Figure 61 presents the variation of mixture ratio as a function of electrical pulse width for Engine Number 1. This is typical of all engines. The mixture ratio decreases as the pulse widths become smaller due primarily to valve mismatch, valve pressure drops and the higher oxidizer vapor pressure. The characteristic performance variation as a function of propellant (oxidizer) temperature is shown in Figure 64. The same trend is noted with fuel temperatures. The effect of propellant temperature on performance was first noted during the PFRT program.

The PFRT Final Report-TMC Report A1055 was published on 20 September 1965.

C. PTC

In the fall of 1964, North American, Grumman. and NASA were considering a design change in the spacecraft in the interest of weight reduction. This change involved going from an active thermal control achieved by means of a liquid glycol loop which NAA had in their modules to a passive thermal control system to be achieved by inherent design characteristics. Grumman did not have active thermal control on the Lunar Module and did not want the weight penalty associated with such a system. Based on this, TMC was directed to initiate studies, analysis, design changes and subsequent development tests to determine the necessary changes to the engine to incorporate the concept of Passive Thermal Control in the spacecraft.

Concurrent with the conduct of the PFRT Program, the Passive Thermal Control (PTC) development program was being pursued leading toward the design that would go into Qualification.

The study program showed that with relatively minor engine changes, significant improvement could be made in the engine's heat transfer to a space environment. The relative merit of the improvements depended on the passive heat transfer to the engine from the vehicle and the operation modes in which the engine is used.

Enough testing was accomplished as a part of the study to define within a close tolerance the thermal characteristics of the Prequal engine (PFRT) and to predict the thermal characteristics of a PTC engine configuration. Further testing was required to validate some of the values measured in the study and to more fully investigate phenomena that could lead to a much higher resistance and more reliable combustor seal.

CALIBRATION TEST
MEAN STEADY STATE PERFORMANCE SUMMARY

	Thrust - lb.		O/F		I_{sp} - sec.	
	Spec.	Data Mean	Spec.	Data Mean	Spec.	Data Mean
Engine No. I						
Results from first Calibration Test	95 ± 2.5	96.2	2.0 ± 0.05	1.984	272 to 294	286.1
Results from second Calibration Test	95 ± 2.5	96.0	2.0 ± 0.05	1.983	272 to 294	287.1
Engine No. II						
Results from first Calibration Test	95 ± 2.5	95.1	1.95 ± 0.05	1.948	272 to 294	284.1
Results from second Calibration Test	95 ± 2.5	96.8	1.95 ± 0.05	1.940	272 to 294	292.1
Engine No. III						
Results from first Calibration Test	95 ± 2.5	94.1	2.0 ± 0.05	1.979	272 to 294	277.7
Results from second Calibration Test	95 ± 2.5	97.1	2.0 ± 0.05	1.991	272 to 294	289.0

Figure 59

CALIBRATION TEST
INDIVIDUAL STEADY STATE RUN PERFORMANCE SUMMARY

Run	Thrust ~ lb.			Isp ~ sec.			O/F Ratio		
	Eng. I	Eng. II	Eng. III	Eng. I	Eng. II	Eng. III	Eng. I	Eng. II	Eng. III
RESULTS FROM FIRST CALIBRATION TESTS									
1	96.6	96.0	94.2	285.1	286.5	277.8	2.006	1.972	1.978
2	96.2	94.8	93.2	286.1	283.6	275.4	2.000	1.945	1.960
3	96.1	94.1	94.7	286.5	282.6	279.6	1.973	1.927	1.986
4	95.9	95.3	94.2	286.8	283.7	277.9	1.955	1.946	1.990
RESULTS FROM FINAL CALIBRATION TESTS									
1	96.0	96.2	97.0	287.0	290.6	288.1	1.985	1.937	1.998
2	96.1	96.9	97.6	286.6	292.7	290.1	1.984	1.944	1.995
3	96.3	97.1	96.7	287.5	292.8	288.2	1.992	1.939	1.987
4	95.8	96.9	97.2	287.4	292.2	289.4	1.972	1.939	1.984

Figure 60

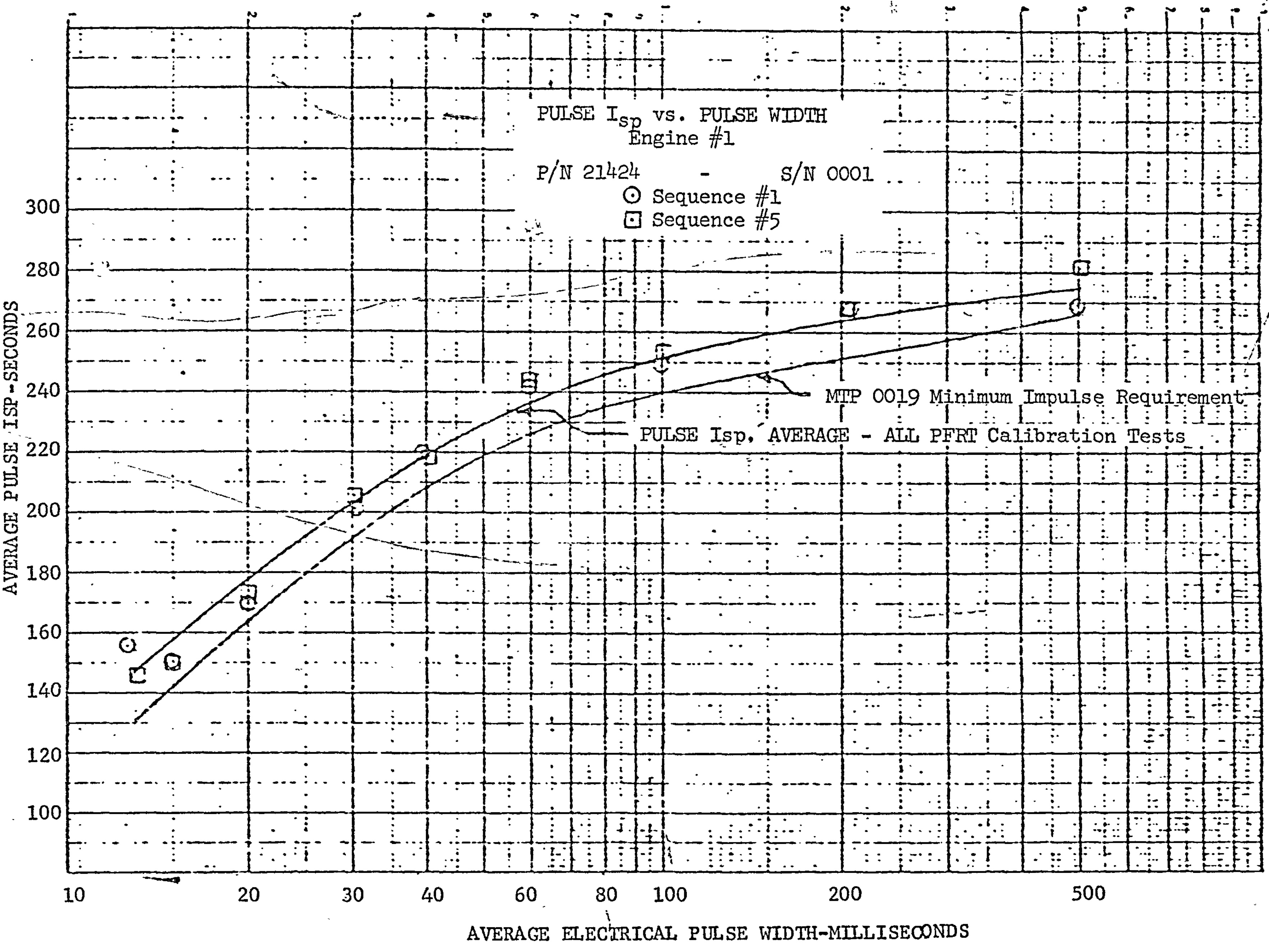

PULSE I_{sp} vs. PULSE WIDTH
Engine #1
P/N 21424 - S/N 0001
⊙ Sequence #1
⊡ Sequence #5
MTP 0019 Minimum Impulse Requirement
PULSE Isp, AVERAGE - ALL PFRT Calibration Tests
AVERAGE PULSE ISP-SECONDS
300
280
260
240
220
200
180
160
140
120
100
10
20
30
40
60
80
100
200
500
AVERAGE ELECTRICAL PULSE WIDTH-MILLISECONDS

Figure 61

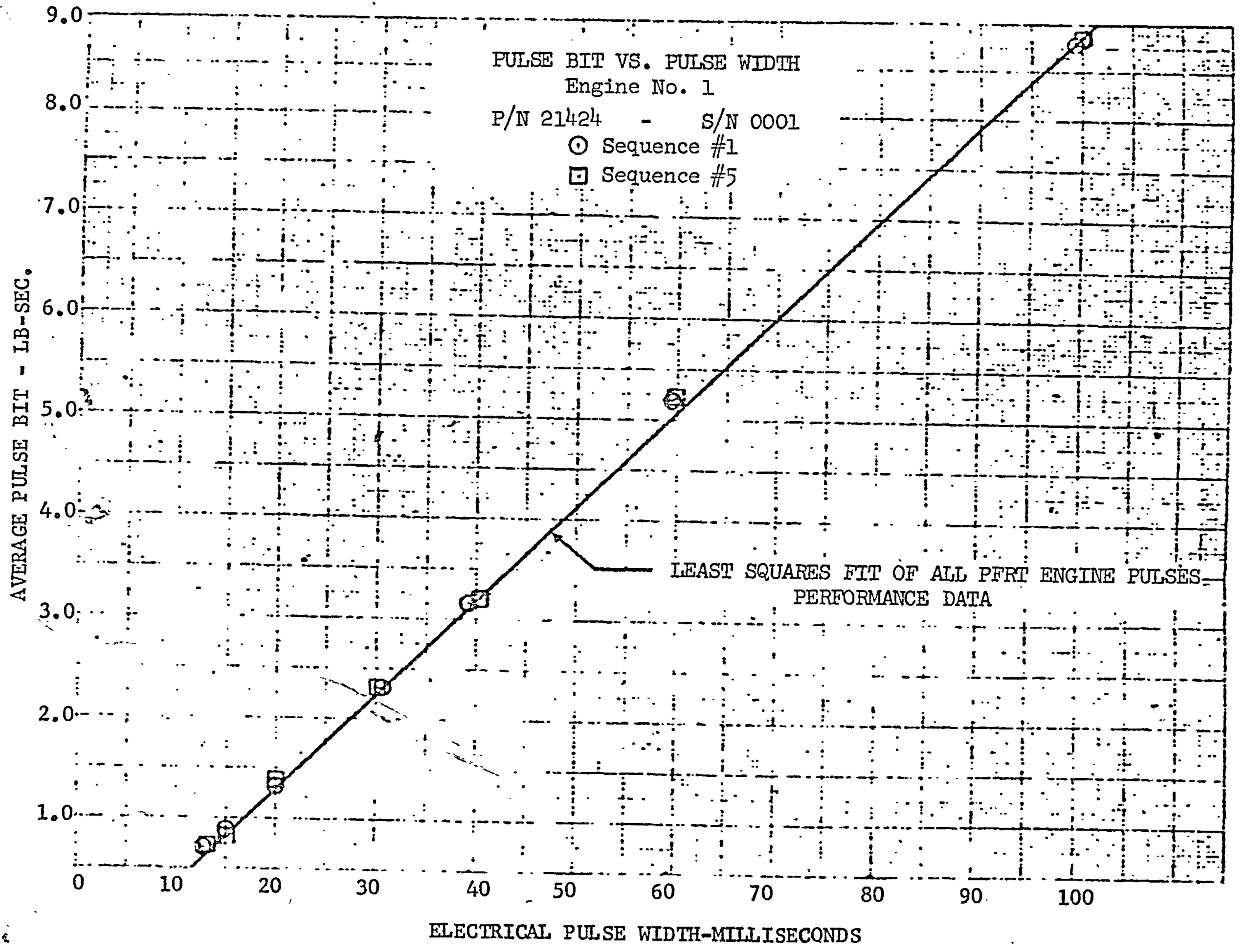

PULSE BIT VS. PULSE WIDTH
Engine No. 1
P/N 21424 - S/N 0001
⊙ Sequence #1
⊡ Sequence #5
LEAST SQUARES FIT OF ALL PFRT ENGINE PULSES PERFORMANCE DATA
AVERAGE PULSE BIT - LB-SEC.
9.0
8.0
7.0
6.0
5.0
4.0
3.0
2.0
1.0
0
10
20
30
40
50
60
70
80
90
100
ELECTRICAL PULSE WIDTH-MILLISECONDS

Figure 62

MIXTURE RATIO VS. ELECTRICAL PULSE WIDTH
ENGINE P/N X21424, S/N 0001

PFRT Engine No. 1

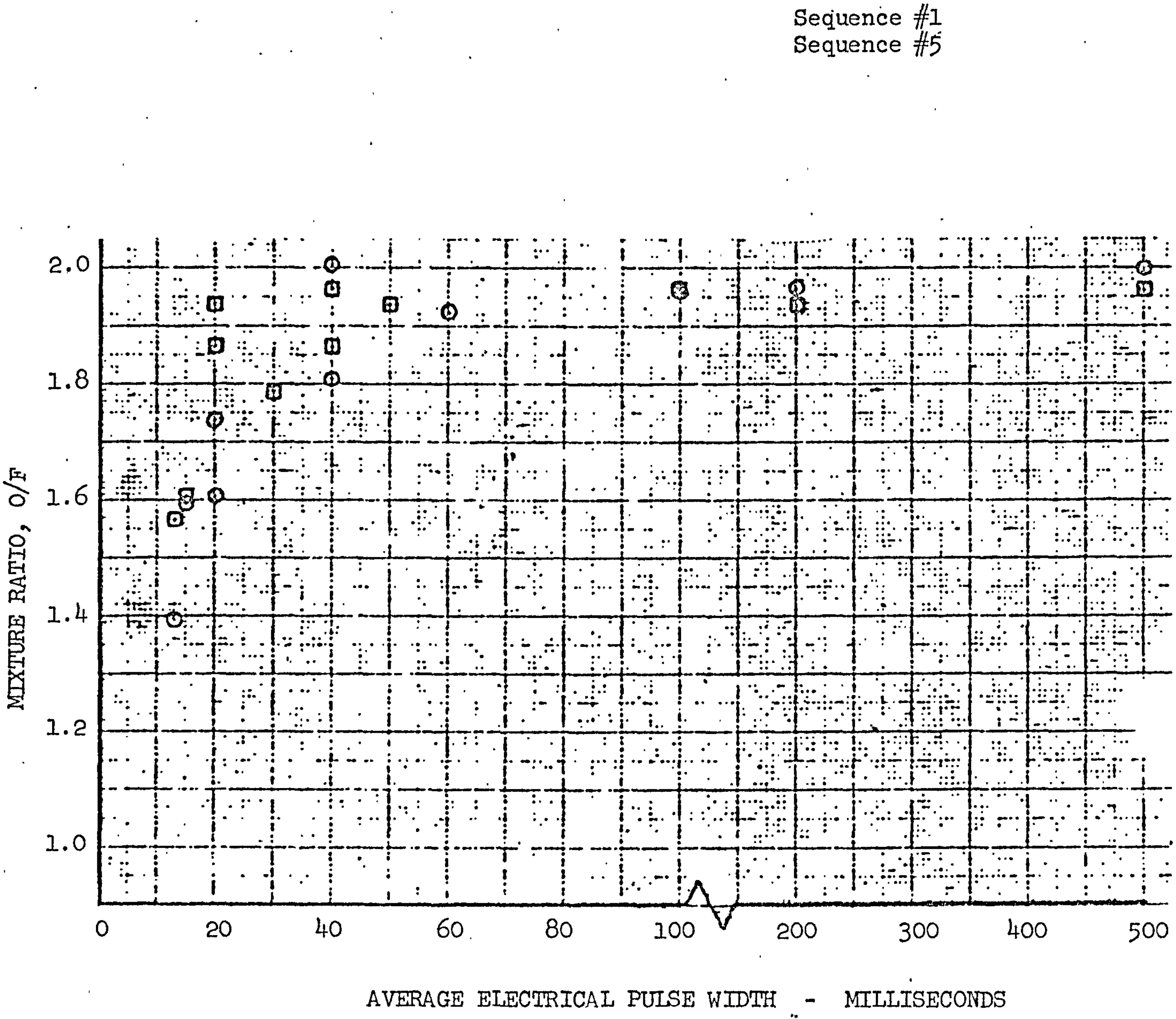


Figure 63

SPECIFIC IMPULSE VS. OXIDIZER TEMPERATURE
Steady State Data Corrected to 170 psia Inlet Pressure
5 Second Data Points Only

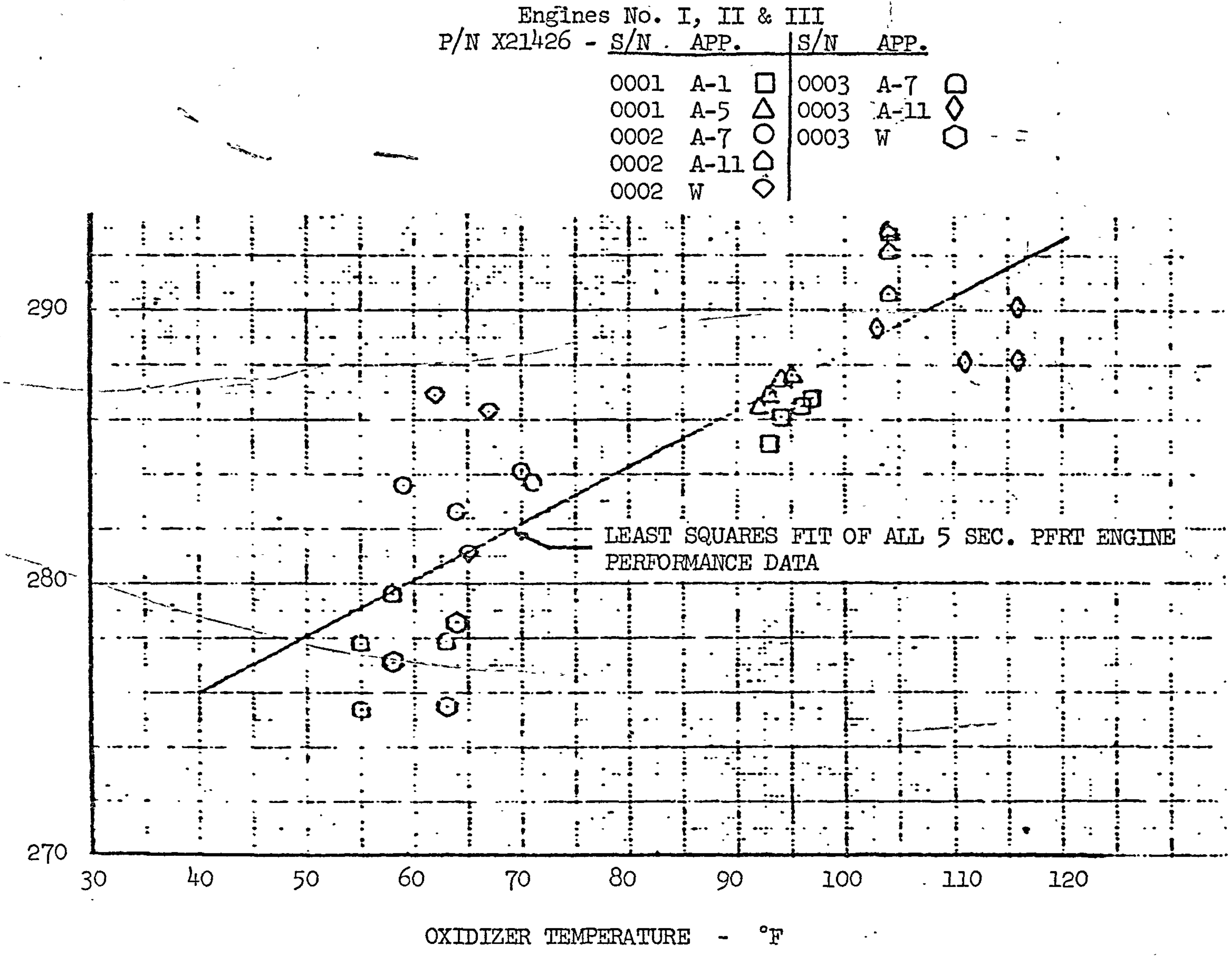


Figure 64

Two areas wherein concentrated effort were expended were in the fuel valve standoff and the injector head to combustor seal.

The Qual configuration program effort was concentrated on the design, analysis and bench testing of the fuel valve standoff to determine the thermal resistances. An analysis of the anticipated qualification design was completed. The minimum thermal resistance of the fuel valve to head was calculated to be 60,600 sec.- °F/BTU based upon the following assumptions:

(a) No internal valve resistances.
(b) No ball seat resistance.
(c) No contact resistances.
(d) Nominal dimensions.

The predicted test value for this configuration was 70,000 sec. - °F/BTU, based on a nominal ball seat resistance of 10,000 sec.-°F/BTU which was calculated from test results. This compares to the requirement of 44,600 sec.- °F/BTU.

Steady state thermal tests at high vacuum conditions with a non-preigniter engine assembly were completed to obtain preliminary data for valve-to-head and head-to-combustor thermal resistances.

The thermal data was analyzed using the following criteria:

(a) The complex, parallel heat transfer paths in the engine valve assembly were simplified to a single path between each valve and the injector head, and a single path between the head and the combustor.

(b) Radiation heat transfer between adjacent components and to the surroundings was neglected.

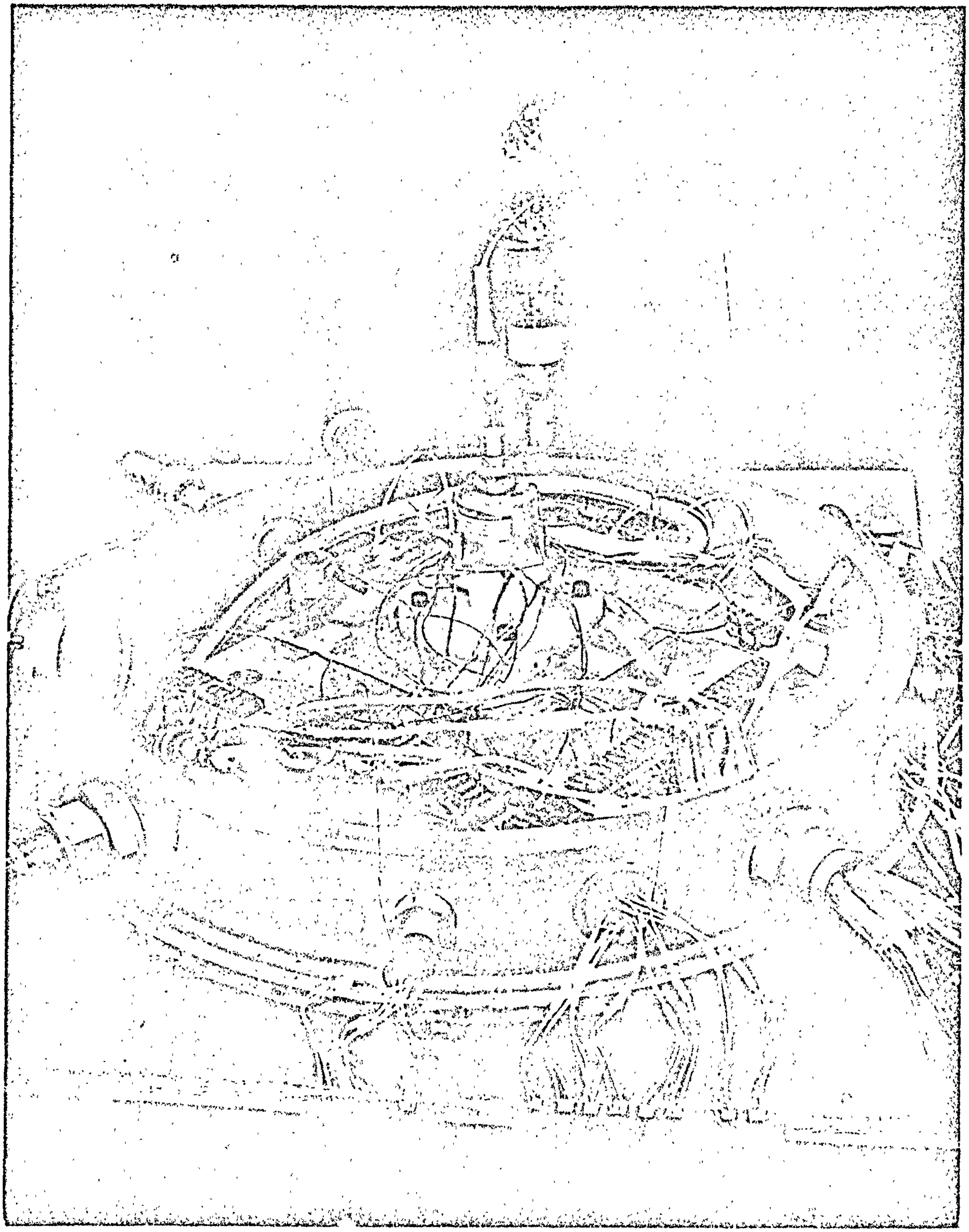
(c) The overall temperatures of the valves were represented by their valve seat temperatures.

(d) The heat flowing in the simplified thermal circuit was identical to the heat generated in the valve electrical coils.

The test sequence and the results of the analysis are summarized in the following table:

Test No.	Omni-Seals (Both Valves)	Oxidizer Valve Phenolic	Remarks	Thermal Resistance (°F-sec/BTU)	
				Oxidizer Valve Seat to Head	Fuel Valve Seat to Head
1	Yes	Yes	Valves empty and seated.	47,800	13,600
2	Yes	Yes	Valves empty and seated.	42,100	12,700
3	Yes	No	Valves empty and seated.	55,100	13,500
4	No	No	Valves empty and seated.	58,300	13,700
5	No	No	Valves empty and seated.	62,100	14,800
6	No	No	Water in valves (oxidizer empty at end)	-	12,950
7	No	No	Water in valves	71,200	12,550

The fuel valve conduction test rig (FVTR) was designed to provide detail thermal resistance data for the fuel valve head connection. Figure 65 shows this test assembly. Tabulated below is a portion of the test results and Figure 66 is a plot of this thermal data using ΔT valve seat to head and electrical input heat as coordinates. Ideal thermal resistance (no variation with temperature, etc.) would yield data which would plot as a straight line through the origin. The slope is equal to the magnitude of the equivalent thermal resistance. The extrapolation of the thermal data for the tests with the short stainless bolts resulted in an equivalent thermal resistance of 56,570 °F-sec/BTU for the low temperature valve seat (35 ± 10°F) and 43,580 °F-sec/BTU for the high temperature valve seat (184 ± 2°F). For the tests with the long titanium bolts, the high temperature valve seat data (185 ± 15°F) yielded a resistance of 52,620 °F-sec/BTU. The cold valve seat data could not be extrapolated in a similar fashion. The extrapolated lines crossing the zero ΔT point



NEG. 6469-1

Fuel Valve Test Rig #2 with Valve Installed

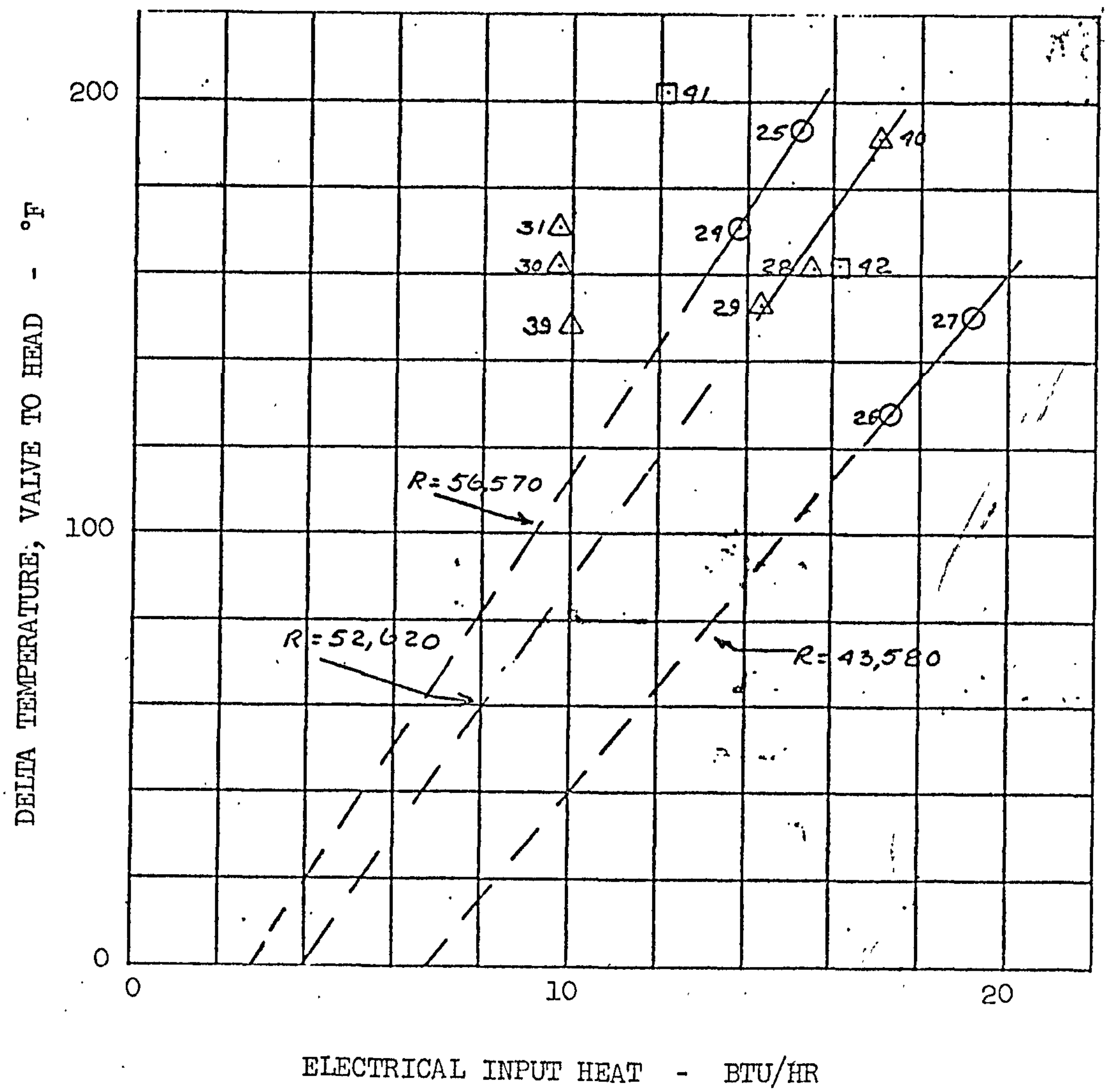
Figure 65

THERMAL RESISTANCE TESTS
ΔT - Valve to Head
vs.
Electrical Input Heat

<u>FVTR #2</u>
⊙ Short S.S. Bolt
△ Titanium Bolt

⊡ <u>FVTR #1</u>

Titanium Bolt (No Phenolic)
(Numbers near symbols denote
run numbers)

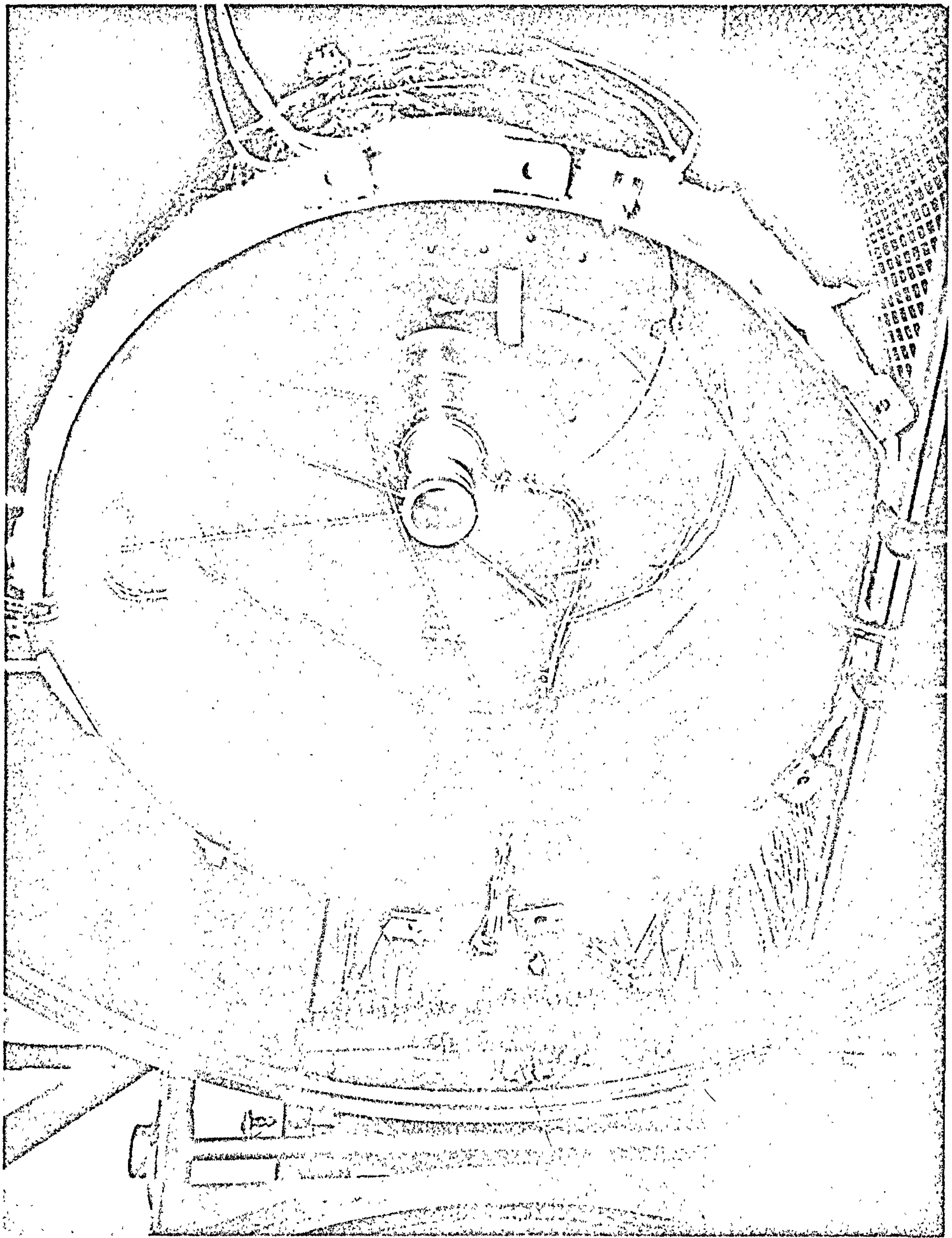


at positive values of heat input can be accounted for as heat leakage to the surroundings.

RESULTS OF FVTR NO. S1 AND 2 THERMAL TESTS

Run No.	Head Temp (°F)	T Valve Seat to Head (°F)	T Valve Ears to Seat (°F)	T Int.Fluid to Seat	Electrical Input Heat (BTU) Hr.	Equiv.Thermal Res. Valve Seat to Head (°F-sec) BTU
24	-145	171	2	19	13.8	
25	-148	193	0	24	15.2	
26	57	128	0	21	17.3	56,570 Short SS
27	31	151	0	24	19.2	43,580 Bolt
28	39	164	9	21	15.5	52,620 FVTR
29	35	153	9	21	14.3	#2
30	-125	162	14	23	9.7	Long TC
31	-131	171	13	23	9.7	Bolt
39	-114	148	13	19	10.0	
40	-23	192	12	21	17.0	52,620
41	-165	202	18	18	12.1	FVTR
42	3	162	16	16	16.1	#1 Long TC
						Bolt, no
						phenolic

A cold wall space simulation facility was fabricated for use in the passive thermal control test program. This facility had the capability of a vacuum of 7×10^{-4} mm of Hg, wall temperatures approximating liquid nitrogen temperature and black space environment with an ϵ of 70.9. The facility is shown in Figure 67. Figure 68 presents the data obtained for steady state temperature distribution and steady state heat loss for several configurations. Heater blankets were installed on valves to provide heat input and chamber configurations and seal materials were changed so that varying rate of steady heat loss was obtained. This data was used to calculate the thermal resistance between the valves and the injector head. The results of this analysis showed the oxidizer valve to head resistance to be 44,000 °F-sec/BTU and the fuel valve to head resistance to be 30,000 °F-sec/BTU for the particular configuration tested (short stainless steel bolts, then phenolic spacer and fuel valve standoff with 90% of the thermal resistance of the Qual standoff). Calculated values for the same configuration were 30,000 °F-sec/BTU for the fuel and 46,000 °F-sec/BTU for the oxidizer valve at 70°F, thus showing good agreement.



NEG. T11215-3

Space Simulation Facility

Figure 67

SPACE SIMULATOR COLD SOAK TESTING

Building 32

Parameter			Test 2	Test 3		Test 4		Test 5
			12-30-64	Run #1 1-5-65	Run #2 1-6-65	Run #1 1-9-65	Run #2 1-11-65	1-13-65
	Hardware Description	Head Seal Chamber Bell	T-11042 Pyrolytic Graphite Fired > 800 sec. Smooth L605-Fired	T-11042 Pyrolytic Graphite Un-Fired Un-Fired Ribbed L-605		T-11043 L-605 Un-Fired Not Connected		T-11043 L-605 Fired Chamber Not Connected
	Heat Input BTU/HR	Fuel	10.6	7.9	8.06	4.84	5.83	7.57
		Ox	7.8	5.6	5.72	2.27	4.55	5.92
		Total	18.4	13.5	13.78	7.11	10.38	13.49
	Calculated Heat Loss	Temp	Tfoil = -170°F	-160°F	-167°F	-156°F	-137°F	-129°F
		q	q = 1.65	1.9	1.73	2.0	2.63	3.1
	Chamber Heat Loss	BTU/HR	16.7	11.6	12.05	5.11	7.75	10.4
STEADY STATE TEMPERATURES	FUEL VALVE	Under Heater	N.G.	+54.5°F	+58.0°F	+8.0°F	+55.0°F	+53.0°F
		Flange	+40°F	+47	+49	+3.0	+48	+45
		Seat	+31	+39	+41	-2.5	+43	+38
		Armature	+47	+47.5	+50	+3.0	+50	+47
	OX VALVE	Under Heater	+55	+53.5	+56	-11	+55	+55
		Flange	+45	+47	+49	-13.5	+50	+49
		Seat	+39	+42.5	+44	-15.5	+47	+44
		Armature	+48	+48	+50	-13.5	+51	+50
	HEAD		-43	-12.5	-12	-30	+11	- 5
	CHAMBER	Moly Flange	-63	-24	-22.5	-33	+ 8	-10
		Throat	-75	-30	-29	-38	+ 3	-16
	BELL	Nut	-81	-37.5	-37	-	-	-
		Mid-Point	-162	-128	-128	-226	-210	-236
		End	-198	-145	-145	-228	-212	-238

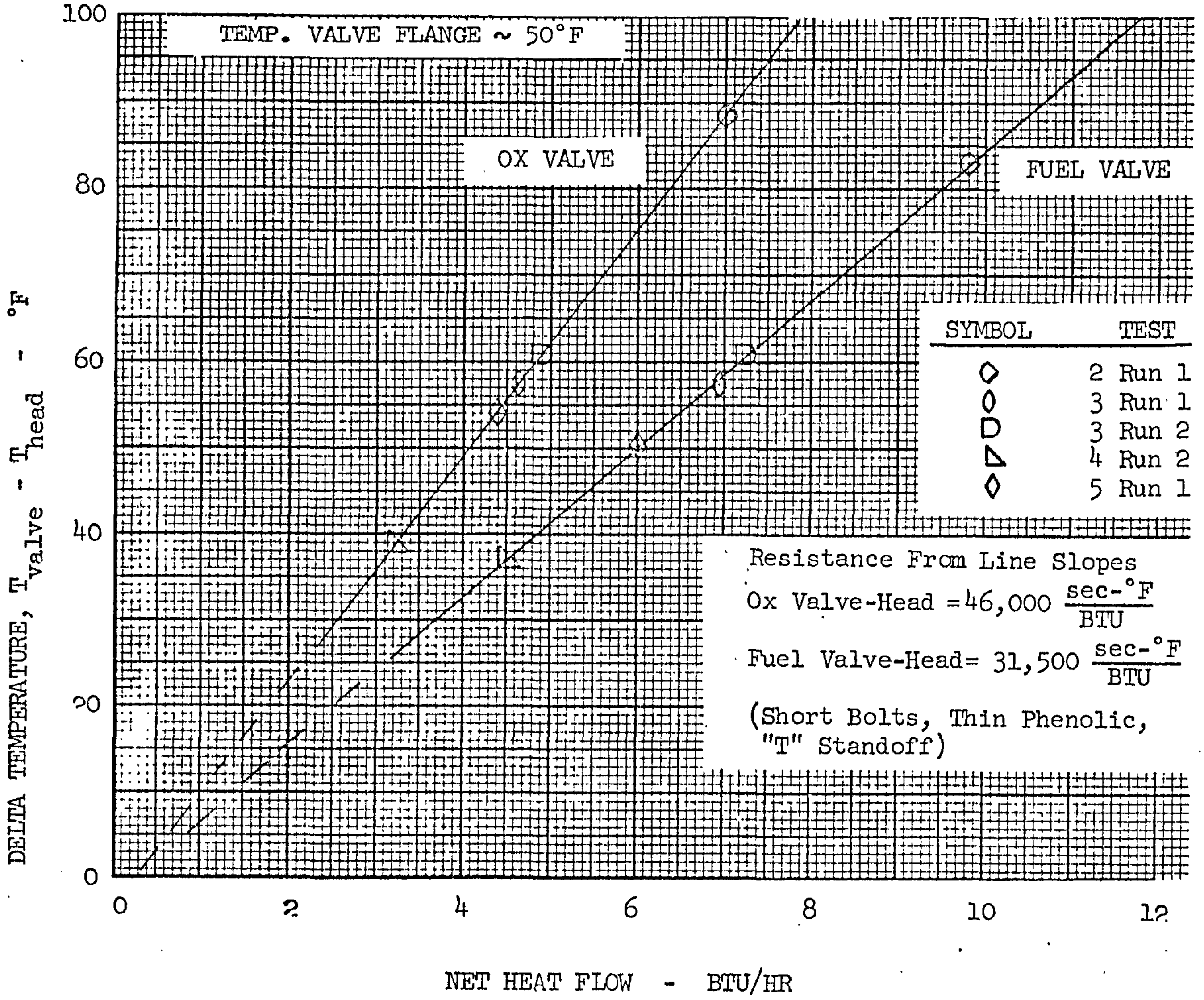
Inherent errors in the measurement of temperature will cause the resistance values to have some variations. The amount of heat loss is dependent upon the assumed emissivity, the effective area, and the internal conduction paths to the outer foil of the insulation. Every effort was made to make the heat loss a minimum. However, significant heat loss did occur as evidenced by the outer foil temperatures listed in Figure 68. The heat loss was calculated based upon an emittance area factor of 0.15 and the outer foil temperature in the standard radiation heat transfer equation. The surface area was determined to be about 1.5 to 1.75 square feet and emissivities from 0.085 to about 0.1 were assumed. These values of emissivity are reasonable for much handled aluminum foil. That the losses are approximately correct is indicated by the intersection point of the net heat flow versus ΔT, (Figure 69). The resulting variation of thermal resistance with average temperature of the standoff is shown by Figure 70. The resistance decreases with average temperature because of the increase of thermal conductivity of the materials with temperature.

In conjunction with the fuel valve standoff design activity, a combustor to injector head seal design configuration was being pursued in an effort to obtain a design which would increase the resistance. Pyrolytic graphite material, asbestos phenolic and the PFRT design L605 seal were being considered. Structural tests were conducted on the material. Random vibration tests were performed on engines with pyrolytic graphite seals without any structural damage. Compression tests performed on the pyrolytic graphite revealed ultimate stress loadings of 30,000 to 36,000 psi. Engine ignition tests resulted in failure of the pyrolytic graphite seal. The asbestos phenolic seal withstood compressive loads to above 100,000 psi without any damage, however, the seal was not effective as a pressure seal because of high leakage rates. Based on these results, the L605 seal was retained which has a thermal resistance of approximately 1800°F-sec/BTU compared to specified design criteria of 18,000 sec.°F/BTU.

In early December 1964, TMC was directed by North American to conduct a valve heater program. This program had as its objective to evaluate the possibility of attaching thermostatically controlled heaters on each valve. The heaters were required to operate at 21 volts dc and to supply valve temperatures of 50°F or less, 8.0 ± 0.8 BTU/Hr. to the oxidizer valve and 12.0 ± 1.2 BTU/Hr. to the fuel valve and were to be thermostatically controlled "on" at fuel valve temperatures of 60°F or less and "off" at fuel valve temperatures of 100°F or greater. This program was discontinued in late January 1965 and no design changes were incorporated as a result of it.

In late January 1965 and early February 1965, the PTC program was brought to a stop and Figure 71 shows the PFRT/AF009 configuration and Figure 72 shows the PTC incorporated changes for the Qual configuration.

SPACE SIMULATION TESTING
Tests 1-5 Building 32



Resistance From Line Slopes

Ox Valve-Head = 46,000 $\frac{\text{sec-}^\circ\text{F}}{\text{BTU}}$

Fuel Valve-Head= 31,500 $\frac{\text{sec-}^\circ\text{F}}{\text{BTU}}$

(Short Bolts, Thin Phenolic, "T" Standoff)

Figure 69

THERMAL RESISTANCE VS. AVERAGE TEMPERATURE

Valve Temperature Constant at ~ 50°F
Space Simulator Tests 2 through 5
Qual Type Oxidizer Valve Configuration
Short Bolts, Thin Phenolic
T Standoff - Fuel Valve Configuration

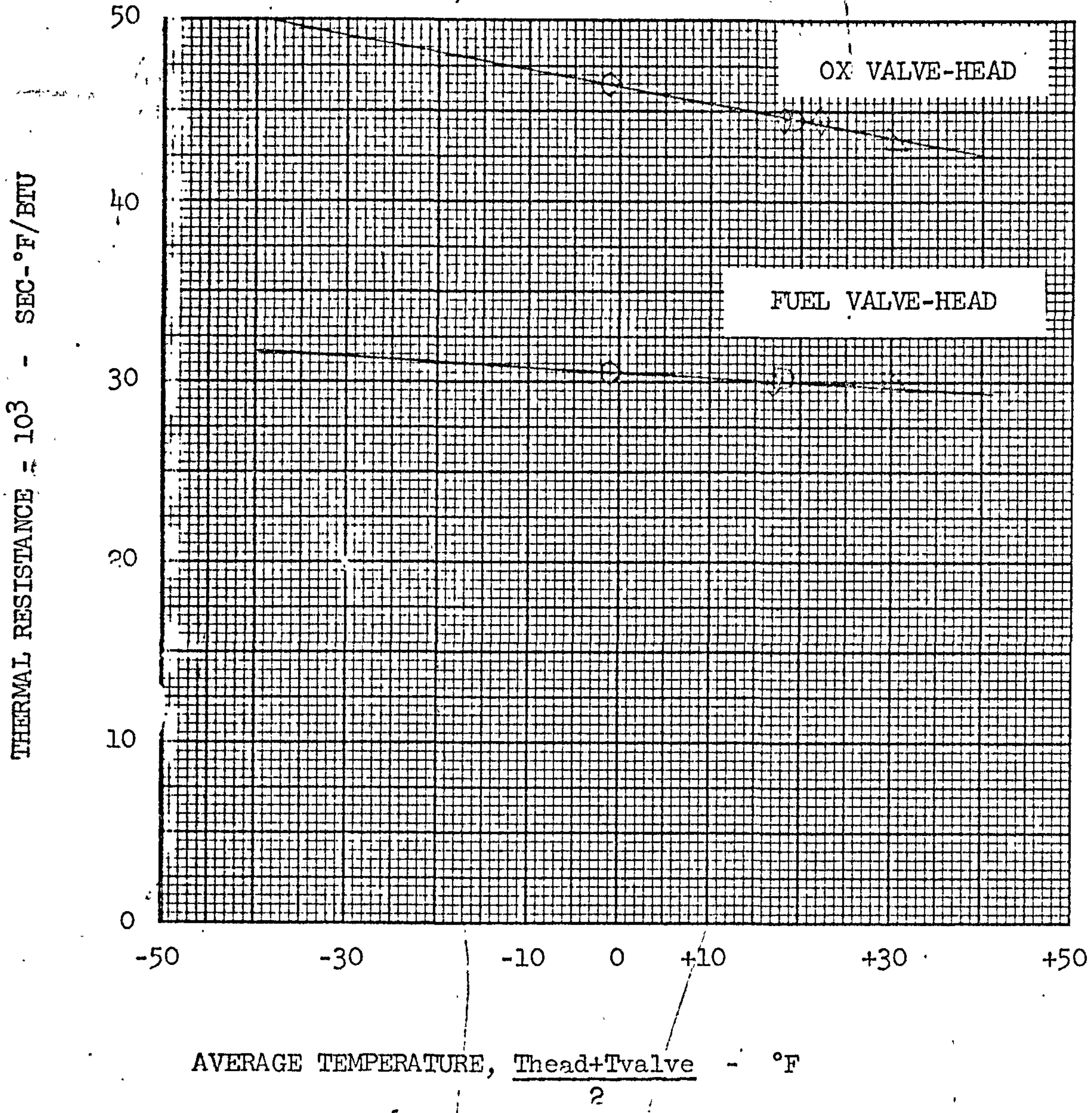


Figure 70

PFRT CONFIGURATION

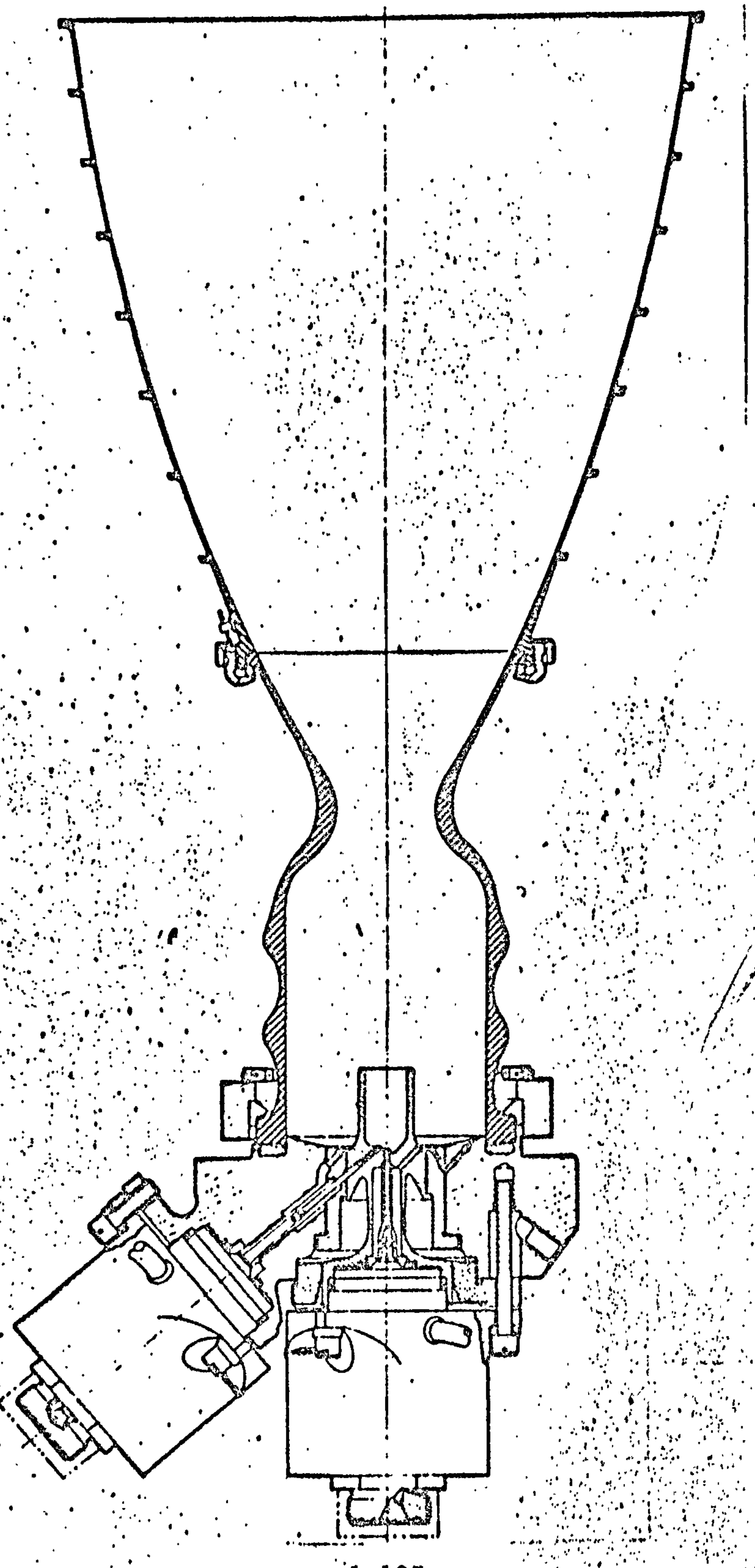


Figure 71

QUALIFICATION CONFIGURATION

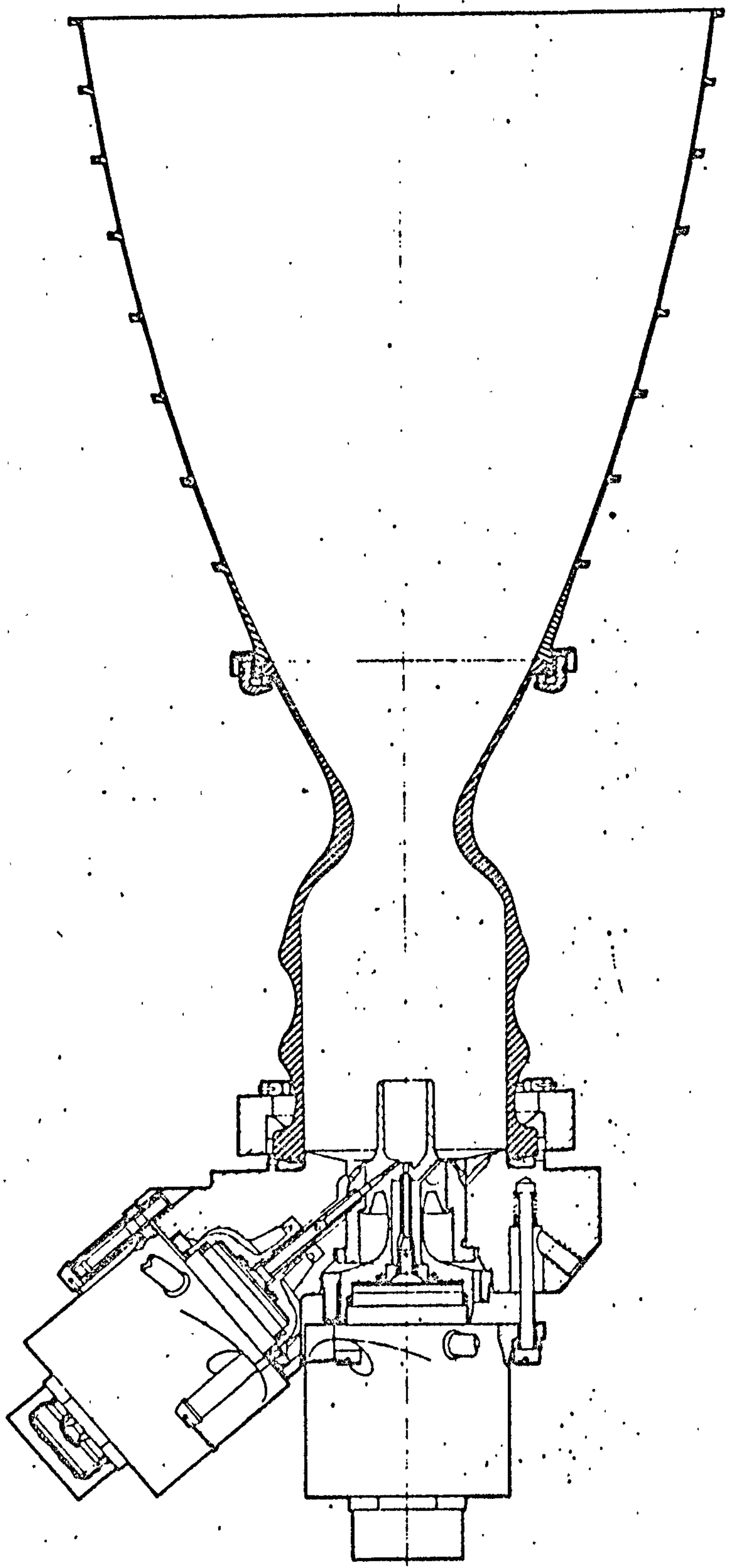


Figure 72

Basic differences are:

1. Fuel Valve Standoff

   (a) Valve on A286 standoff rather than aluminum head.

   (b) Valves secured to head by long 6 AL-4V titanium screws rather than short Cres 17-4 PH screws (17-4 PH fasteners were deleted from Apollo acceptable materials list). A286 thermal spacers were used to increase the heat flux path.

2. Oxidizer Valve Standoff

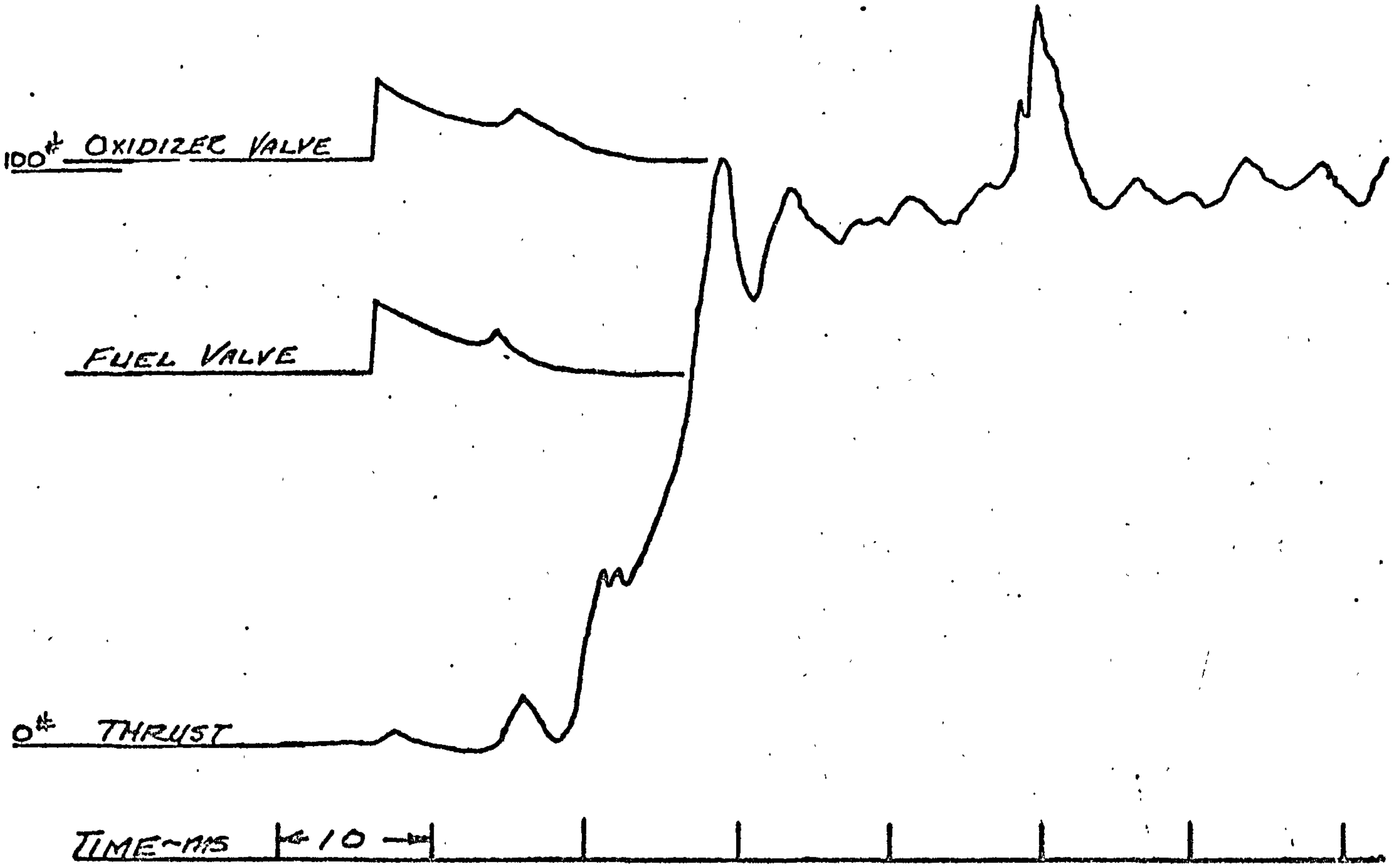
   (a) 17-4 PH screw replaced by Cres A286 screws (to eliminate 17-4 PH fasteners) and 17-7 PH spacer replaced by Cres A286 spacer.

Test data values showed an equivalent fuel valve to head thermal resistance of 52,600°F-sec/BTU compared to specified value of 44,600. The oxidizer valve to head resistance is approximately 61,000°F-sec/BTU.

During testing, in the fall of 1964, oscillations had been noted with the engines. This instability had a frequency of 350 to 400 cps and the cause was traced to entrained N_2 gas bubbles in the propellant. Elimination of entrained gas during acceptance testing was accomplished by installing floats in the propellant run tanks. The floats reduced the area exposed to the pressurizing gas to a minimum value (clearance between the float and the tank wall). In addition, experimental and analytical efforts have shown that nitrogen dissolved in the propellants can have a degrading effect on engine start performance, both immediately after priming the engine and later during engine operation. Experimental evidence indicated that dissolved nitrogen should have no effect on steady state performance; however, there is a discernable improvement in apparent preigniter operation during a run series when degassed propellants are used to fire the engine. Full scale engine tests were conducted using propellants which were degassed by heating to 100°F with the pulse tanks vented to very nearly atmospheric. Runs with both degassed and gassy propellants indicated essentially no difference in engine steady state performance level as a result of the degassing operation. Ignition characteristics appeared to be somewhat more clearly discernable in the oscillograph thrust signature when degassed propellants were used than when gassy propellants were used. Figures 73 and 74 show the comparative start characteristics as a function of propellant condition.

Preigniter ignition occurs at about the same time with either gassy or degassed propellants; however, again the preigniter characteristic is clearer when degassed propellants are used. Samples taken of gassy and degassed propellant

IGNITION CHARACTERISTICS



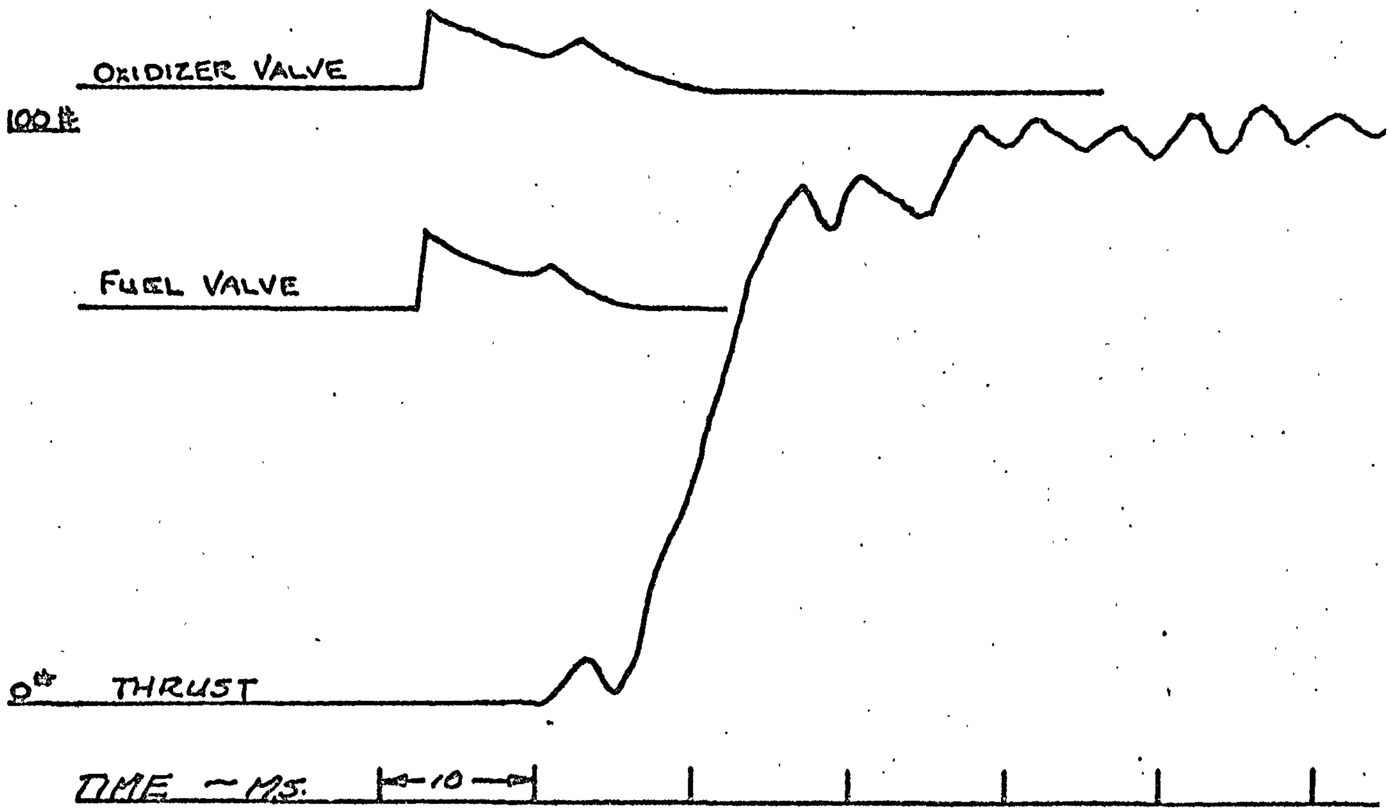
ENGINE T-10670, S/N 002-3
TESTED 23 SEPT. 64 RUN 8459

DEGASSED PROPELLANTS USED
FOR THIS TEST

NOTE THAT PREIGNITER OPERATION
IS CLEARLY DISTINCT

Figure 73

IGNITION CHARACTERISTICS



ENGINE T-10670, S/N 002-3
TESTED 23 SEPT 64 — RUN 8436

GASSY PROPELLANTS USED FOR THIS TEST

NOTE THAT PREIGNITER OPERATION IS INDISTINCT

Figure 74

were subjected to physical and chemical analysis. Degassing does not appear to affect either propellant density or chemical properties in that all samples tested to date meet the propellant specifications.

An additional phenomenon noted during this time period was the I_{sp} decreasing approximately 3 seconds after about 20 seconds of run time. This effect had not been noted in non-preigniter engines. One possible explanation is that there is a slight change in weeper flow characteristics resulting in lower effective performance from this portion of the propellant; however, this possibility was later found to be invalid.

Another possible explanation seems to be related to variations in the amount of fuel flow to the preigniter. The effect of a variation in preigniter fuel flow upon engine performance is twofold. First, variations in preigniter fuel flow rates cause inverse variations in the main doublet fuel flow rates. This causes variations in the doublet Rupe number and accompanying performance changes. Second, preigniter engines are not as efficient as non-preigniter engines, which indicates that the propellant is not burned as efficiently in the preigniter itself as it is when injected through the main doublets. Therefore, changes in the amount of propellant in the preigniter cause changes in the overall level of engine efficiency.

A quantitative measure of the two above effects obtained from the engine test results show that an increase in preigniter fuel flow of 5% of the total fuel (i.e., say an increase from 12% to 17% of the total fuel flow) will lead to a doublet Rupe number change sufficient to cause a 2-second decrease in I_{sp}. In addition, if the performance levels of preigniter engines are compared to those of non-preigniter engines, and if the difference is attributed to the inefficiency of combustion in the preigniter, then an increase in preigniter fuel flow of 5% of the total fuel flow will cause a decrease in engine I_{sp} of 1.3 seconds. Thus, the total decrease in specific impulse will be 2.0 ± 1.3 = 3.3 seconds.

The numerical values given above assume no change in the oxidizer flow rate to the preigniter. If the preigniter oxidizer flow rate should increase, the reduction in doublet Rupe number would cause an I_{sp} increase which would approximately balance the decrease in I_{sp} caused by the increased amount of propellant in the preigniter. If the preigniter oxidizer flow rate should decrease, the doublet efficiency would decrease, but this would be balanced by the increase in efficiency due to less propellant in the preigniter. Thus, it appears that the performance effects of preigniter oxidizer flow variations may be negligible. No conclusive evidence has been made of the above possible cause.

Although not believed to be related to the above phenomenon, the effect on performance due to thermal growth and manufacturing tolerances of small nozzle throat area changes on rocket engine parameters is summarized below for throat area changes of ± 2%.

A differential analysis was used to calculate the relative change from assumed reference engine performance parameters for nozzle throat area changes as great as ± 4%. Figure 75 shows the percentage change in thrust chamber pressure, specific impulse and propellant flows for throat area changes from 0 to ± 4%. Characteristic exhaust velocity (C*) and O/F were assumed to be constants for this analysis.

Summary of Results for Change of ± 2%

For $\frac{dAt}{At} = \pm .02$

$$\frac{dI_{sp}}{I_{sp}} = \mp .0018$$

$$\frac{dF}{F} = \pm .0054$$

$$\frac{dPc}{P_c} \doteq \mp .0128$$

$$\frac{d\dot{w}_p}{\dot{w}_p} \doteq \frac{d\dot{w}_f}{\dot{w}_f} = \frac{d\dot{w}_e}{\dot{w}_o} = \pm .0072$$

$\frac{dc^*}{c^*} = 0$ Assumption

$\frac{dO/F}{O/F} = 0$ Assumption

Assumed Reference Engine Performance

$I_{sp_{vac}}$ = 290 seconds

F_{vac} = 95 pounds

P_c = 90 psia

$\dot{w}_p$ = .3276

O/F = 2.00

$P_{mo} = P_{mf}$ = 170 psia

Nominal throat area = 0.5915 in.^{2}

ROCKET ENGINE PERFORMANCE PARAMETER CHANGES RESULTING FROM SMALL NOZZLE THROAT AREA CHANGES

$C^* = \text{Constant}$

$O/F = \text{Constant}$

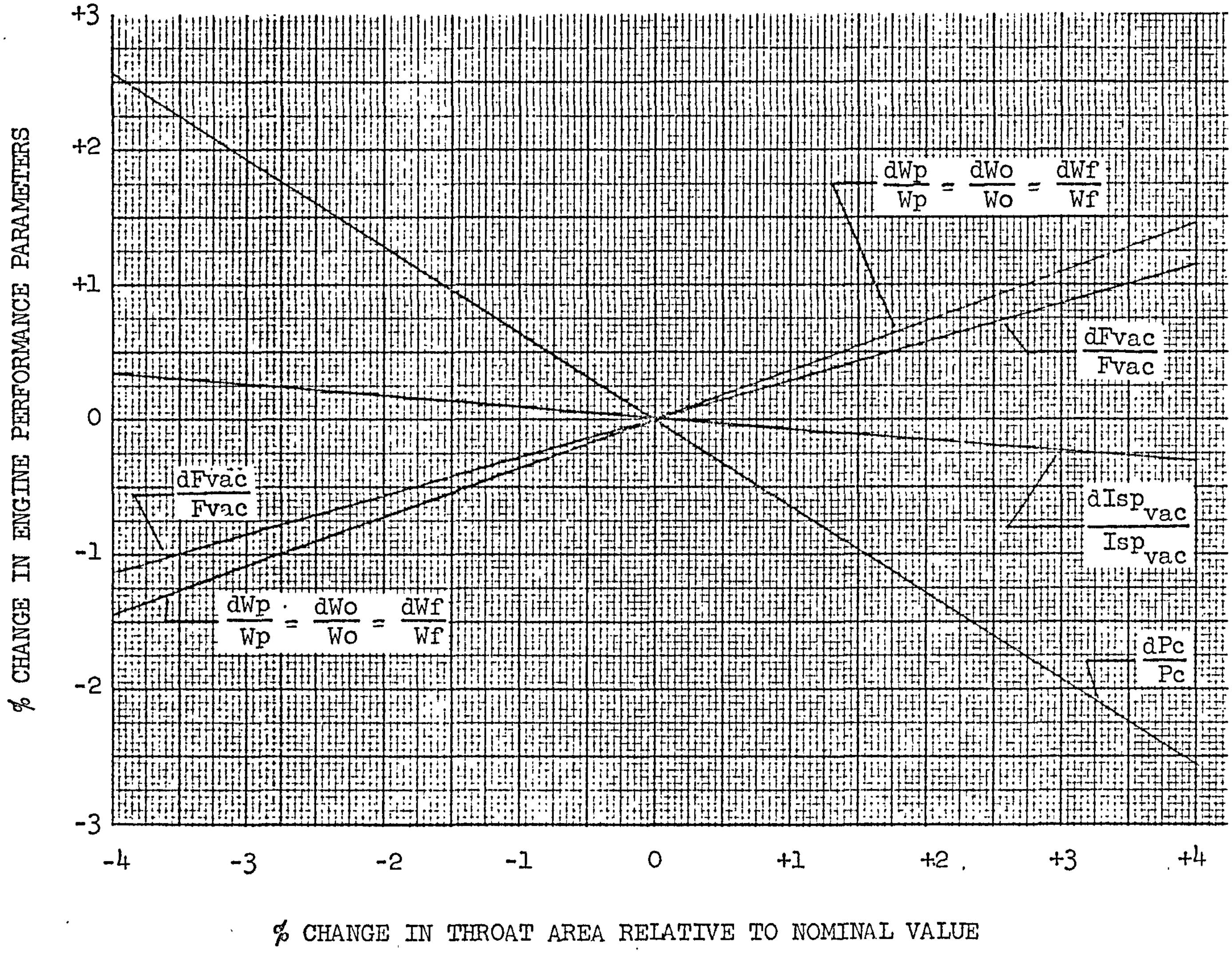


Figure 75

During early 1965, tests were conducted on engines wherein slight design modifications had been made in an attempt to increase performance. One such change was to increase oxidizer doublet size from .0355 to 0.0365 diameter. This change resulted in increased performance both with hot and cold propellant, as compared to PFRT, AFRM 009 and PTC engine experience. The results are shown on Figure 76. Each of these engines had the same injector configuration except for T11605, S/N 0001-1 which had fuel hole diameters smaller by 0.001 inch compared to the other engines. The above change was not incorporated into the Qual design. It was believed that additional testing would be required to substantiate the change and it was decided to remain with the doublet sizes as used in the PFRT design.

A low pressure drop preigniter tube for 100 lb. thrust was developed and incorporated into the design such that the Qual engines would have 100 lb. thrust rather than the nominal 95 lb. thrust of the PFRT engine.

In late March and early April 1965, Design Substantiation tests were initiated on the engine and the solenoid valve. This hardware was identical in every way to the Qualification engine design. Where possible, Qual procedures etc., were used - the explicit objectives were:

1. Steady State Engine Performance
   - (a) Demonstrate the effect of the supply propellant temperature upon thrust, I_{sp} and O/F.
   - (b) Demonstrate the effect of run time upon thrust, I_{sp} and O/F.
   - (c) Demonstrate the effect of variable O/F upon the engine specific impulse.
   - (d) Demonstrate safe engine operating of off design O/F's with hot propellants.
   - (e) Define transient and steady state engine operating temperatures at design and off O/F design conditions.
2. Pulse Performance
   - (a) Define the pulse specific impulse, total impulse and O/F for various pulse on times as a function of:
     1. Valve full open mismatch
     2. Propellant temperature
     3. Valve voltage
     4. Off time

PERFORMANCE WITH LARGER OXIDIZER DOUBLET

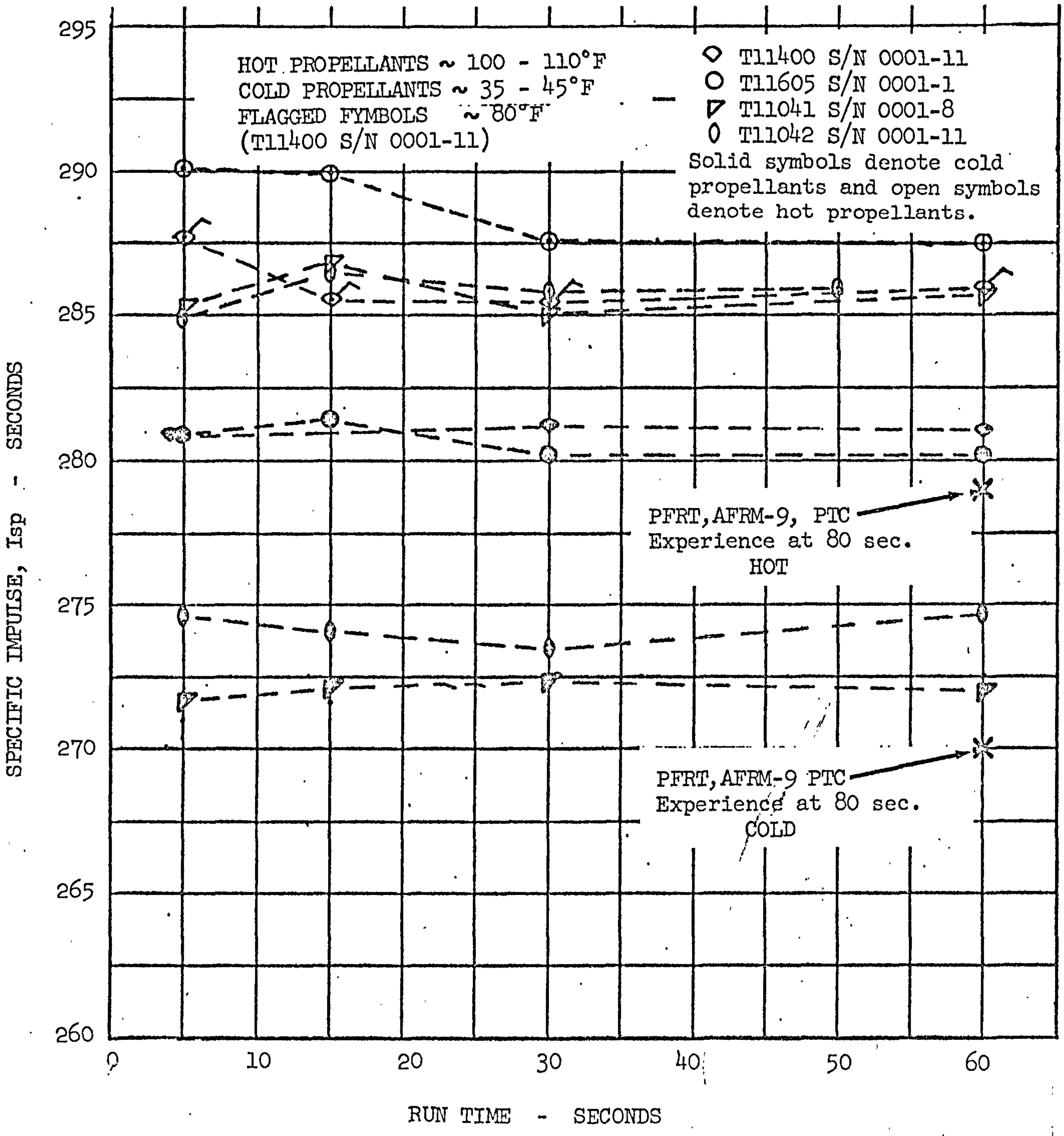


Figure 76

3. Ignition Behavior

   (a) Determine the maximum ignition pressure levels obtained at various valve mismatches and 170 psia manifold pressures when the engine component temperatures are the minimum expected (Qual design).

   (b) Determine the maximum ignition pressure levels resulting from pulses conducted with off design manifold pressures at a design valve mismatch.

The Design Substantiation Test of the propellant injector valves of the Apollo Qualification engine configuration were conducted principally to provide assurance of Qualification test capability on a component level. Since the predominant valve failure mode had been seal leakage, the objectives were directed toward demonstrating sufficient seal life. Based on considerable evidence that the seal design utilized on this valve is subject to wear and this wear is accelerated by various environments, the tests were designed to expose the valves to a cyclic pattern of these environments which would simulate a seal wear condition as severe as that which will be experienced during engine qualification.

The valve configuration being tested (P/N 228198 and 228199) was identical to the AFRM 009 design (P/N 228109 and 228111) on a geometrical basis. There were detail differences, the most significant of which are listed below. These changes are all related to seat assembly and its detail parts. The differences are:

1. The seat assembly detail parts (insert, seal, and seat)

   (a) A change in the insert detail to provide a recess for the Teflon seal (this recess was previously in the seat detail).

   (b) An insignificant geometric change to the Teflon seal.

   (c) A change in the seat detail compatible with the seal recess in the insert.

2. The seat assembly details of the Qual configuration were assembled by cooling the insert and seal with liquid nitrogen and squeezing them into the heated seat. With the AFRM 009 assembly, the insert was cooled with N_2 and the seat was heated and the Teflon seal remained at room temperature just prior to squeezing into an assembly.

c. A closer fit between the valve body and the seat assembly register diameters has reduced possible body seat eccentricity.

There was no evidence of a change in the Teflon physical characteristics at room temperature due to the liquid nitrogen soak prior to assembly.

Based on results from that portion of the Apollo valve DST program completed, it became apparent that the 228198 and 228199 valve assemblies had the sealing capability necessary for successful engine Qualification test performance. Four valves of the present configuration (two fuel and two oxidizer) were subjected to over 12,000 cycles of operation (including 10,000 cycles with propellant), and exposed to 200°F and 20°F (and actuated at these extremes). During this exposure, each valve was measured at least 18 times for leakage (a total of 72 individual measurements); and of these measurements, 7 were made during or after the propellant exposure. During only 7 of these measurements (total for all valves) was any measurable leakage detected, and the maximum periodic leakage checks were made and plastic molds made of the seats before and after 10,000 cycles in propellants.

The original Design Substantiation Program (DST) was expanded to include off design ignition tests and minimum and maximum temperature environmental tests to the Qual test conditions and included a second engine, as well as the conduct of a Pre-Qualification test phase to check out procedures, test setups, etc. Steady state and pulse runs showed satisfactory operation, and ignition tests showed the engine did not generate ignition spikes of sufficient level to cause hardware degradation. Maximum pressures encountered were 2900 psi which occurred under off design conditions of a 10 ms oxidizer lead.

Temperature tests did indicate design deficiencies. Under hot propellant conditions (injector head approximately 300°F, valves 175°F, and propellants 100°F), oxidizer flow rate decreases were noted. Subsequent teardown revealed that the seat had degraded to the point where the flow was being affected. During ignition testing, a fuel valve exhibited excessive leakage.

Design Substantiation and Pre-Qualification tests showed the need for the following changes:

1. Valve seat modification to provide better resistance to high temperature.

2. Propellant degassing to be accomplished frequently during qualification testing.

3. Changes to test setups to accomplish better temperature conditioning and temperature measurement.

As a result of the DST valve seat problem, a high temperature propellant exposure valve program was started to determine the operational modes and mechanism under which such leakage would occur. The seat assemblies used were:

(a) Double angle, glass filled Teflon design which failed test.

(b) Double angle, pure TFE grade 7 Teflon.

(c) Single angle, pure TFE grade 7 Teflon (design developed by TMC IR&D). The single and double angle configurations are depicted in Figure 77.

The tests performed were:

(a) Valve actuation at high and low propellant temperatures with steady state pressure drops measured before and after cyclic test.

(b) Vibration to the boost levels of the present requirements.

(c) Regular GN_2 leakage checks, in which the valve was totally submerged in water (ambient leak check) or alcohol (+35°F leak check) and the valve oriented to permit direct observation of the poppet seat area. The valves were disassembled and visually inspected at regular intervals to assess the effects of the tests.

During the early phases of testing, the single angle TFE Teflon seat showed considerable promise, while the double angle seat (with both the filled and unfilled Teflon) indicated sufficient Teflon flow, in contrast to the single angle seat, to preclude further evaluation. Consequently, the major part of the investigation was directed toward ensuring the suitability of the single angle, pure TFE Teflon seat for qualification testing.

This program was, in reality, a continuation of the valve seat development program, undertaken in late 1964. Engine tests were also conducted to evaluate the new single angle, pure Teflon seat configuration. The valve seats, during hot calibrations, were heated to 175°F and no valve seat leakage occurred.

During mid-Summer 1965, helium saturation tests were conducted on the engine to determine engine operation stability when operated with propellants saturated with gaseous helium. Early results indicated the engine operated with stability under all conditions, where helium was used as the pressurant and saturating gas. A crosscheck, using gaseous nitrogen as the pressurant and saturating gas, resulted in definite unstable engine operation. However, subsequent testing with helium showed thrust oscillations occurring intermittently at a frequency

VALVE SEAT DESIGNS

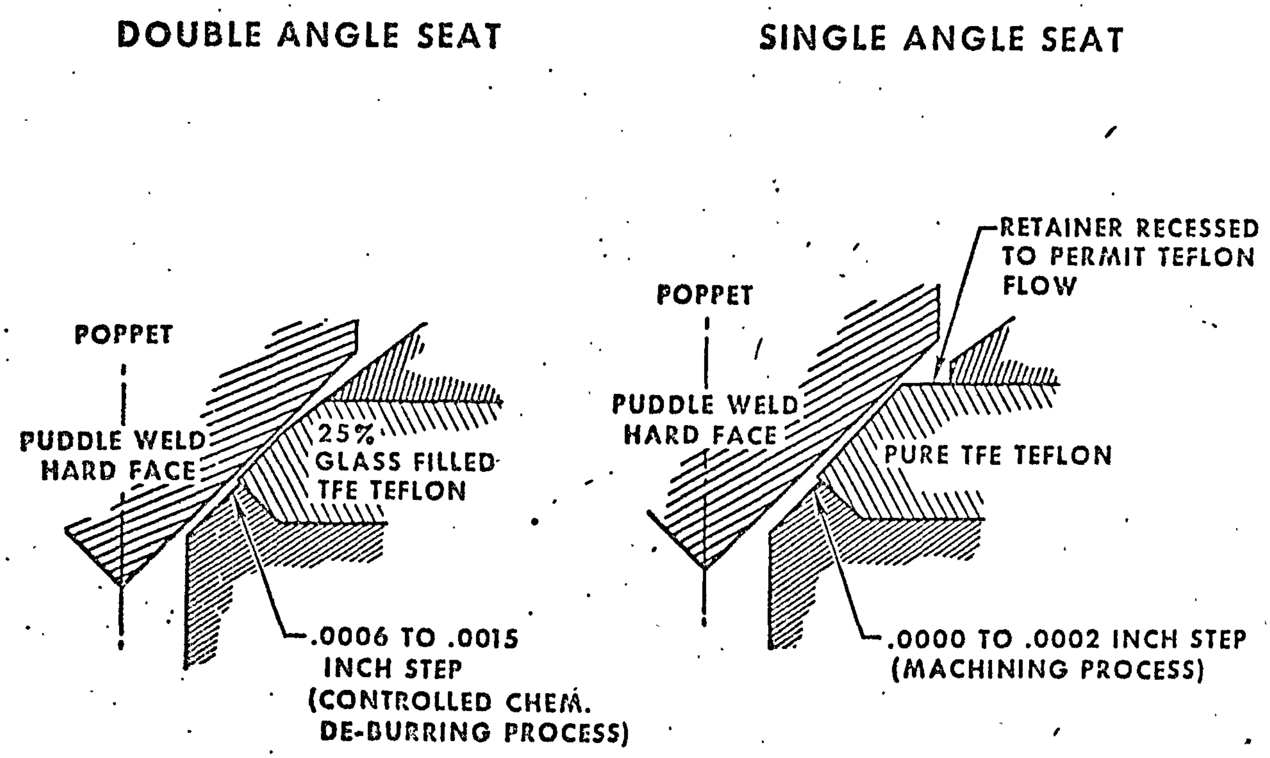


Figure 77

of 300 to 350 cps and an amplitude of P_c up to ± 50 psi. Measurable specific impulse does not appear to be affected and ignition overpressures increased in magnitude.

Although it became definite that helium saturation had to be considered in the design of the engine, it was decided to conduct the Qualification Test Program with "degassed" (low saturation level) propellants pressurized with helium for the initial calibration tests on three engines and run one Qualification engine through its complete test matrix with "degassed" propellants. Helium effects tests were to be conducted in parallel with Qualification.

V. QUALIFICATION

North American approvals of Qualification drawings, Qualification Test Procedures, and End Item Acceptance Test Procedures were received on August 13, 1965. Selection and instrumentation of Qualification Engines #1 and #2 were accomplished on August 14, 1965, Engines #3, #4, and #5 were selected on August 18, 26, and 30, 1965, respectively.

The Qualification engines, TMC P/N 228687 were:

Engine Number	Serial Number	Letter Description
1	0002	A
2	0009	B
3	0005	C
4	0013	D
5	0017	E

The first of the five engines began Qualification testing on August 16, 1965.

Marquardt Report A1051-A, dated 9 August 1965, is the Qualification Test Plan and Procedures. All requirements for inspection and testing were in accordance with NASA NPC 200-2.

A. Qualification Test

Figure 78 presents the Qualification test matrix for the five engines, and the following presents a brief description of each test.

Calibration Test - Three types of calibration tests were conducted on the Qualification engines; these were with "HOT", "AMBIENT", and "COLD" propellant temperatures. The tests consisted of several series of pulses, plus steady state firing, in order to determine engine performance characteristics. The "HOT" calibration was conducted with 100°F propellant temperatures and valve voltages of 24 volts. The "AMBIENT" calibration was conducted with 75°F propellant temperatures and valve voltages of 27 volts dc. The "COLD" calibration was conducted with 40°F propellant temperatures and valve voltages of 30 volts dc.

Transportation Shock Test - The engine, packed in the shipping container, was subjected to a terminal peak sawtooth shock of 30 g's with a period of 11 milliseconds in each direction of the three orthogonal axes.

QUALIFICATION TEST PROGRAM

Test No.	Test Name	Engine Number				
		1	2	3	4	5
1	Calibration Test (Ambient)	X	X	X	X	X
2	Shock (Transportation)	X				
3	Vibration (Transportation)	X				
4	Humidity	X				
5	Salt, Fog	X				
6	Static Load	X				
7	Boost Vibration		X		X	
8	Boost and Space Vibration	X		X		X
9	Electrical and Structural Integrity	X	X	X	X	X
10	Calibration Test (Hot)	X				
11	Calibration Test (Cold)		X			
12	Mission Simulation (Hot)	X				
13	Mission Simulation (Cold)		X			
14	Pulse Operation Survey	X	X	X	X	X
15	Calibration Test (Ambient)			X		
16	Mission Calibration (Ambient)			X		
17	Orbit Retrograde		X		X	
18	Direct Coil Duty Cycle	D X	X	X	X	X
19	Calibration (Ambient)	X	X	X	X	X
20	Electrical and Structural Integrity	X	X	X	X	X
	Total valve actuations	18,491	15,057	17,507	11,463	16,100
	Total burn time (seconds)	1,490	1,758	1,467	1,393	1,405

Propellants (Controlled Saturation)
Fuel - A50
Oxidizer - N_2O_4
Pressurant - Helium

Figure 78

Transportation Vibration Test - The engine, packed in the shipping container, was subjected to a sinusoidal vibration for 1 hour in each of the three orthogonal axes. The vibration levels were as follows:

Frequency	Level
10 to 27.5 cps	± 1.56 g's.
27.5 to 52 cps	0.043 inch double amplitude.
52 to 500 cps	± 6.0 g's.

Space Vibration Test - The engine was subjected to random vibration levels as might be encountered during operation of the upper stages of a launch vehicle in space. The engine was at the minimum temperature to be expected during a space mission. Vibration was conducted for 10 minutes in each axis with levels as follows:

20 to 100 cps	Linear increase on a log-log scale from 0.003 to 0.15 g^2/cps.
100 to 2000 cps	Constant 0.015 g^2/cps.

Boost Vibration Test - The engine was subjected to random vibration levels as might be encountered during operation of the first stages of a launch vehicle. Engine temperatures were ambient. Test duration was 5 minutes in each axis at the following vibration levels:

10 to 90 cps	0.055 g^2/cps at 10 cps with an increase of 3 db per octave to 0.5 g^2/cps at 90 cps.
90 to 250 cps	Constant at 0.5 g^2/cps.
250 to 2000 cps	0.5 g^2/cps at 250 cps with a decrease of 3 db per octave to 0.06 g^2/cps at 2000 cps.

Static Load Test - The engine was subjected to tests which simulated the expected static air loads during launch. Loads were applied at 2.42 and 7.84 inches from the engine mounting flange. The loads applied were increased to a maximum of 210 and 319 pounds at the two loading points. No engine damage or deformation resulted from the test.

Humidity Test - The engine was subjected to an environment of greater than 95% relative humidity and a temperature in excess of 125°F for 10 days.

Pulse Operation Survey - An engine was subjected to two surveys of 4000 pulses with a range of pulse widths from 10 to 500 milliseconds and at various repetition rates resulting in 10 to 300 millisecond off times.

Salt Fog Test - The engine was subjected to 48 hours of a salt fog environment of Method 509 of MIL-STD-810.

Orbit Retrograde Test - The engine was subjected to a test to demonstrate capability for safe steady state operation for 500 seconds.

Mission Simulation Test - The engine was subjected to a series of 5,650 pulses with a total burn time of 530 seconds to simulate firing sequences to simulate operation during a typical mission. Tests were conducted at "HOT", "AMBIENT", and "COLD" conditions, as in the Calibration Test.

Direct Coil Duty Cycle Test - The engine was operated with 21 and 32 volts dc, using the direct coil, and with static valve inlet pressures of 181 and 250 psia. The engine was tested with pre-firing temperature at ambient and at minimum non-operative levels.

Electrical and Structural Integrity Test - Each Qualification test engine was tested for electrical and structural integrity at the completion of a series of environmental tests, and again at the completion of firing tests. The engine was pressure checked to demonstrate structural integrity. The valves were subjected to a transient voltage spike of 50 volts peak with a pulse width 10 microseconds to demonstrate that they would not be damaged by transient voltage. The valve response and resistance characteristics were measured and compared with characteristics determined during acceptance testing to demonstrate electrical integrity.

On 17 September, approximately one month after the start of the Qualification Program, Qual Engine No. 2 (designated as Engine B), experienced a combustion chamber failure during cold mission testing.

During the conduct of the cold mission test an indication of a large oxidizer leak was observed part way through the run. The test was halted and attempts made to determine the cause of the leak. Pressure checks of the facility did not reveal the cause and the altitude chamber was opened and visual observation of the engine revealed the damage. A failure investigation was promptly initiated and all hot firing testing of Qual engines was suspended. Evaluation of the data during the failure runs indicated that an explosion occurred in the oxidizer passages of the injector head during the sixth pulse of the failure run. This explosion caused severe damage to the oxidizer seat which in turn resulted in a massive oxidizer leak, which continued until the end of the run. As a result of the oxidizer leak, the combustion chamber failed on the 20th pulse due to an ignition pressure spike. All of the above was substantiated by various parameter traces recorded during the run.

During the investigation, tests were conducted on a Development engine to attempt repetition of the Engine B failure. Injector head explosions were generated under the temperature and duty cycle conditions with valves which did not leak. Additional tests demonstrated that the explosions were caused by fuel being retained in the oxidizer propellant passages from the end of one engine firing until the beginning of the next. The presence of this fuel then caused an explosion when the oxidizer was introduced. Tests indicated the occurrence of such explosions are strongly dependent upon engine temperatures, times between engine firings, and engine back pressures. Tests were also conducted with monomethylhydrazine (MMH) and revealed that ignition pressures were much lower than those with Aerozine-50.

The formal Failure/Malfunction Report (FMR 279-110) for the structural failure of Qualification Engine B was submitted to NAA/S&ID on November 5, 1965. The report established the cause of failure as being an explosion in the injector oxidizer passageway which damaged the oxidizer valve seat, resulting in a large oxidizer leak. The oxidizer leak led to combustion chamber overpressures during subsequent pulses, and finally to a chamber failure.

The failure report concluded that the failure mode experienced was caused by operating the engine under improper environmental conditions. It was established by test and analysis that the injector head explosion which led to the failure was caused by the relatively high engine back pressure maintained in the Marquardt Test Cell (i.e., Cell No. 1). It was shown that the explosion would not have occurred in the true space environment.

A program of additional work to understand the transport mechanism for getting fuel into the oxidizer manifold was conducted and any abnormal oxidizer inlet manifold overpressure was classified as a "zot". The results of this program are presented in TMC Report S-483 and discussed in Chapter 3.

Hot firing tests of the Qualification program were resumed on October 28, 1965.

A replacement engine for Engine B was selected and completed ambient calibration tests on November 23, 1965. The engine selected was S/N C049 and was designated as Engine B^1. The taped portion of the cold mission simulation test was conducted in TMC's Cell 9 rather than Cell 1 in order to obtain a closer simulation of space conditions.

Engine A completed final testing on November 22, 1965; Engine C on December 7, 1965; Engine D completed all tests on November 8, 1965; Engine E completed final tests on December 14, 1965, and Engine B^1 completed all tests on December 31, 1965. The Qualification Test Report, TMC Report A1057, was submitted to NAA/S&ID on 17 January 1966. All five engines successfully completed the structural, environmental and firing tests. Figures 78 through 85 present the performance of Engine A during the Qualification test and the average pulse performance for all five engines.

A summary of planned and actual valve actuations and firing times are presented below:

	Valve Actuations		Firing Time - Seconds	
	Planned	Actual	Planned	Actual
Engine No. 1	17,220	18,491	1446	1490.37
Engine No. 2	13,147	15,057	1562	1758.38
Engine No. 3	17,173	17,507	1399	1467.22
Engine No. 4	10,538	11,463	1360	1393.05
Engine No. 5	15,673	16,100	1374	1404.50

In summary, the Apollo SM RCS Engine Qualification Test Program was successfully completed.

All engine environmental tests were successfully completed. They demonstrated capability of the engine to reliably meet the requirements of boost and space vibration, boost air loading, transportation shock and vibration and high humidity and corrosive salt fog exposure as defined in NAA/S&ID Procurement Specification MC 901-0004E.

Six engines were fired during this program. One was damaged due to a facility-induced explosion within the injector. Periodic checks of the other five engines' electrical and structural integrity were made throughout the test program. They demonstrated a high level of consistency of component operational characteristics, with no degradation of engine seals.

The five engines which successfully completed the Qualification Test Program demonstrated greater operational life capability than that required by the test plan.

TOTAL IMPULSE VS. VALVE VOLTAGE
QUALIFICATION TEST
ENGINE NUMBER 1

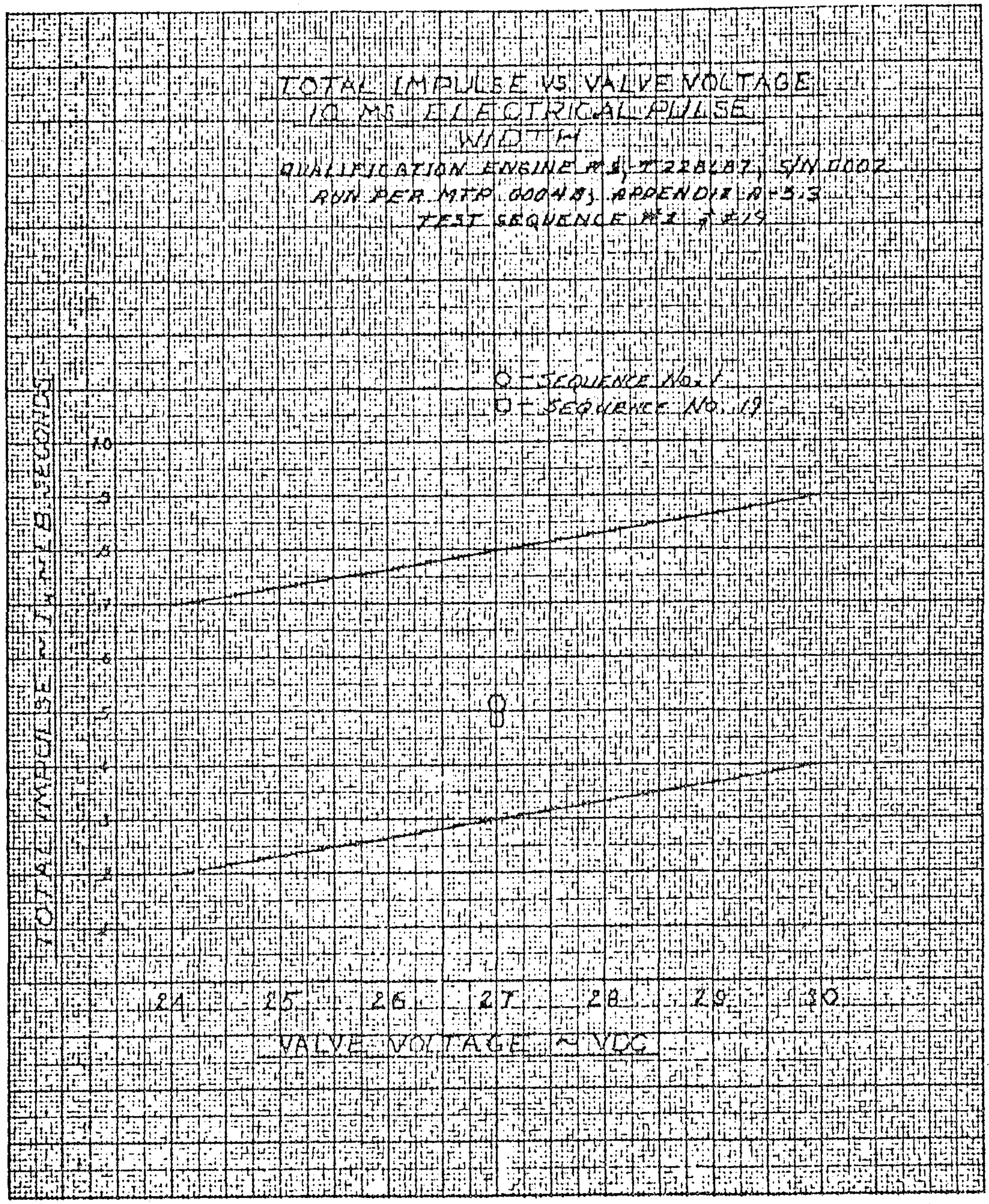


Figure 79

SPECIFIC IMPULSE VS. ELECTRICAL PULSE WIDTH
QUALIFICATION TEST
ENGINE NUMBER 1

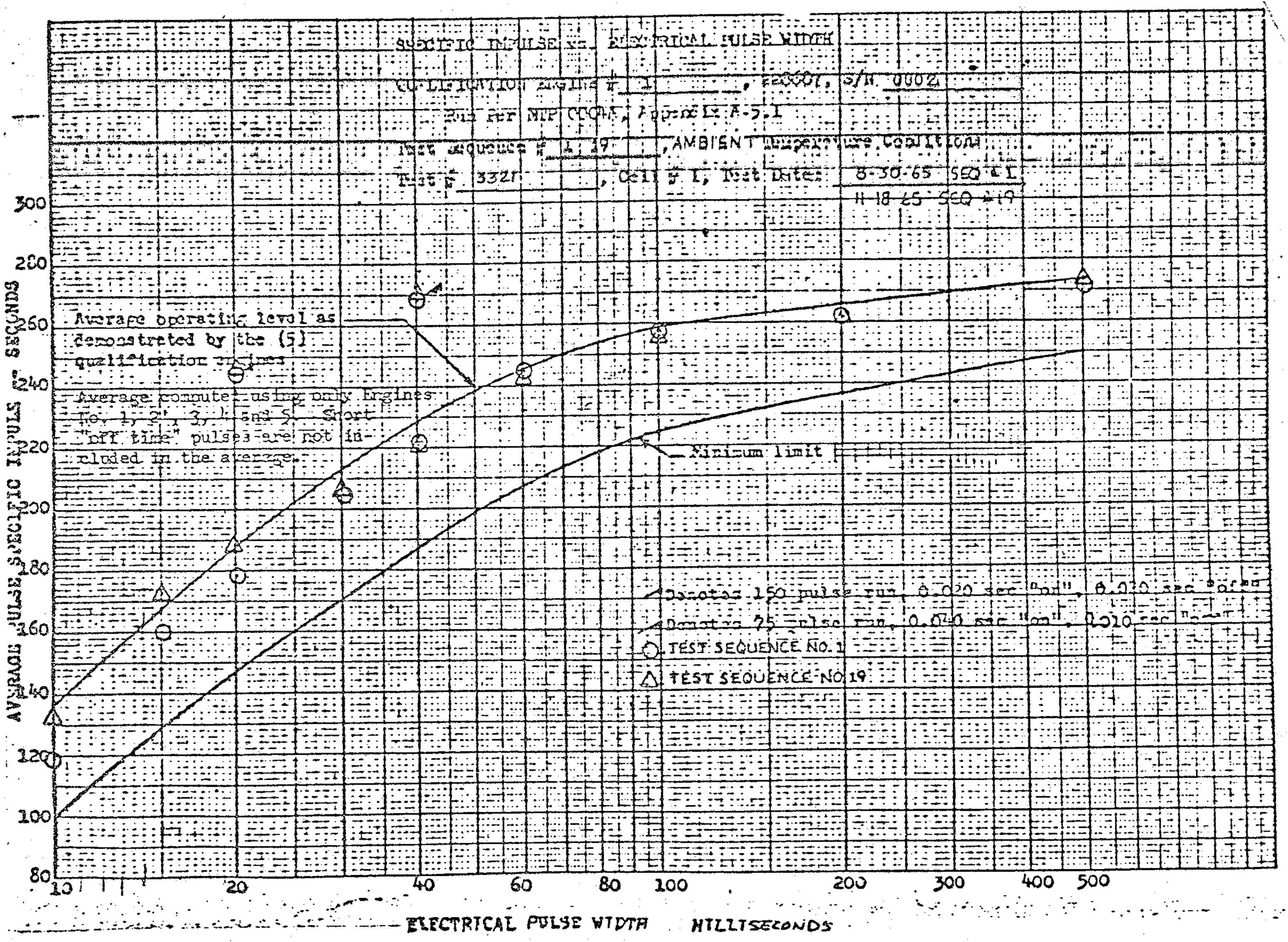
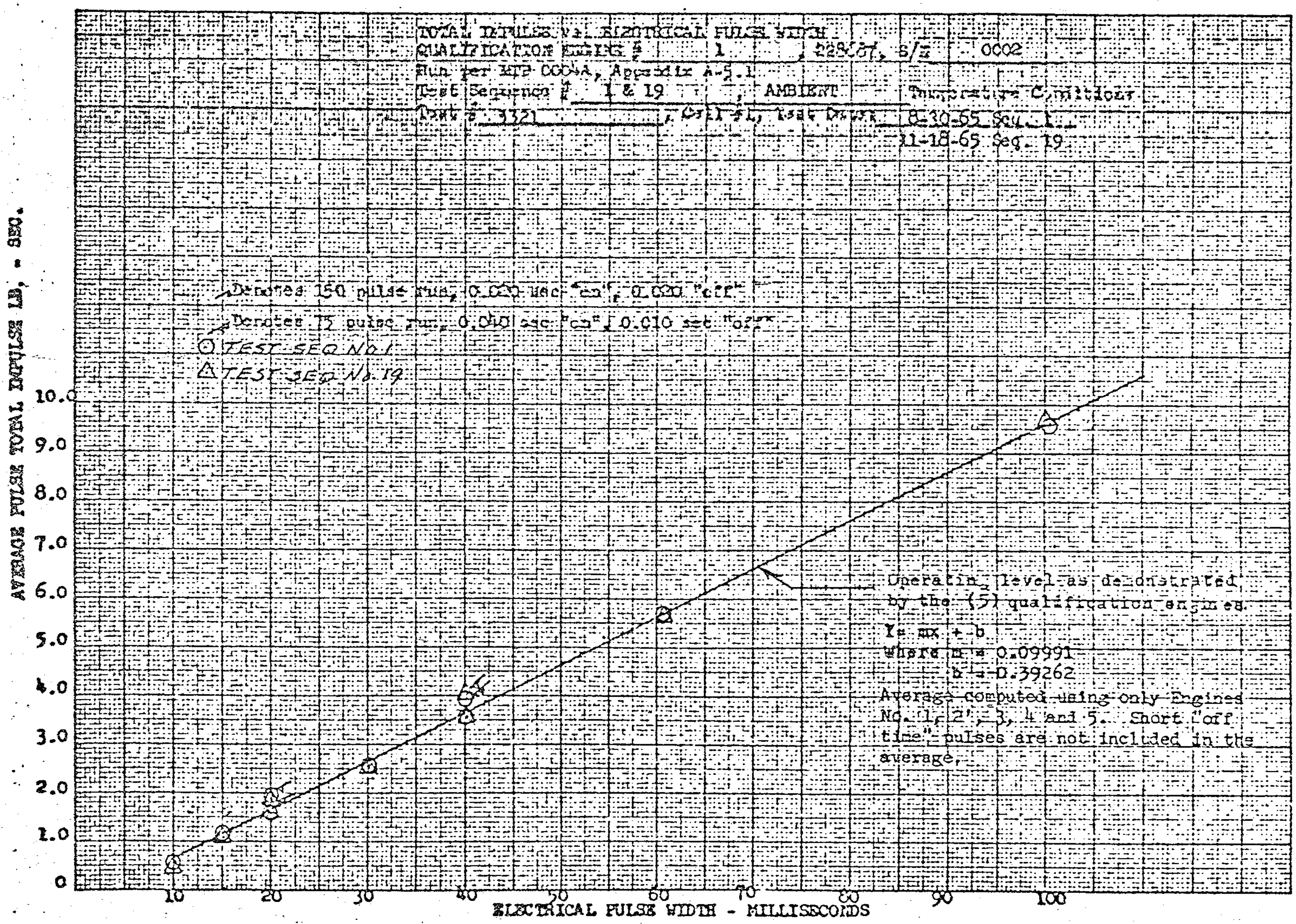


Figure 80

TOTAL IMPULSE VS. ELECTRICAL PULSE WIDTH
QUALIFICATION TEST
ENGINE NUMBER 1



TOTAL IMPULSE vs. ELECTRICAL PULSE WIDTH
QUALIFICATION ENGINE # 1 , 228087, S/N 0002
Run per MTP 0004A, Appendix A-5.1
Test Sequence # 1 & 19 , AMBIENT Temperature Condition
Test # 3321 , Cell #1, Test Dates 8-30-65 Seq. 1
11-18-65 Seq. 19

Denotes 150 pulse run, 0.020 sec "on", 0.020 "off"
Denotes 75 pulse run, 0.040 sec "on", 0.010 sec "off"
⊙ TEST SEQ No 1
△ TEST SEQ No. 19

Operating level as demonstrated by the (5) qualification engines.
Y = mx + b
Where m = 0.09991
b = -0.39262
Average computed using only Engines No. 1, 2', 3, 4 and 5. Short "off time" pulses are not included in the average.

Figure 81

MIXTURE RATIO VS. ELECTRICAL PULSE WIDTH
QUALIFICATION TEST
ENGINE NUMBER 1

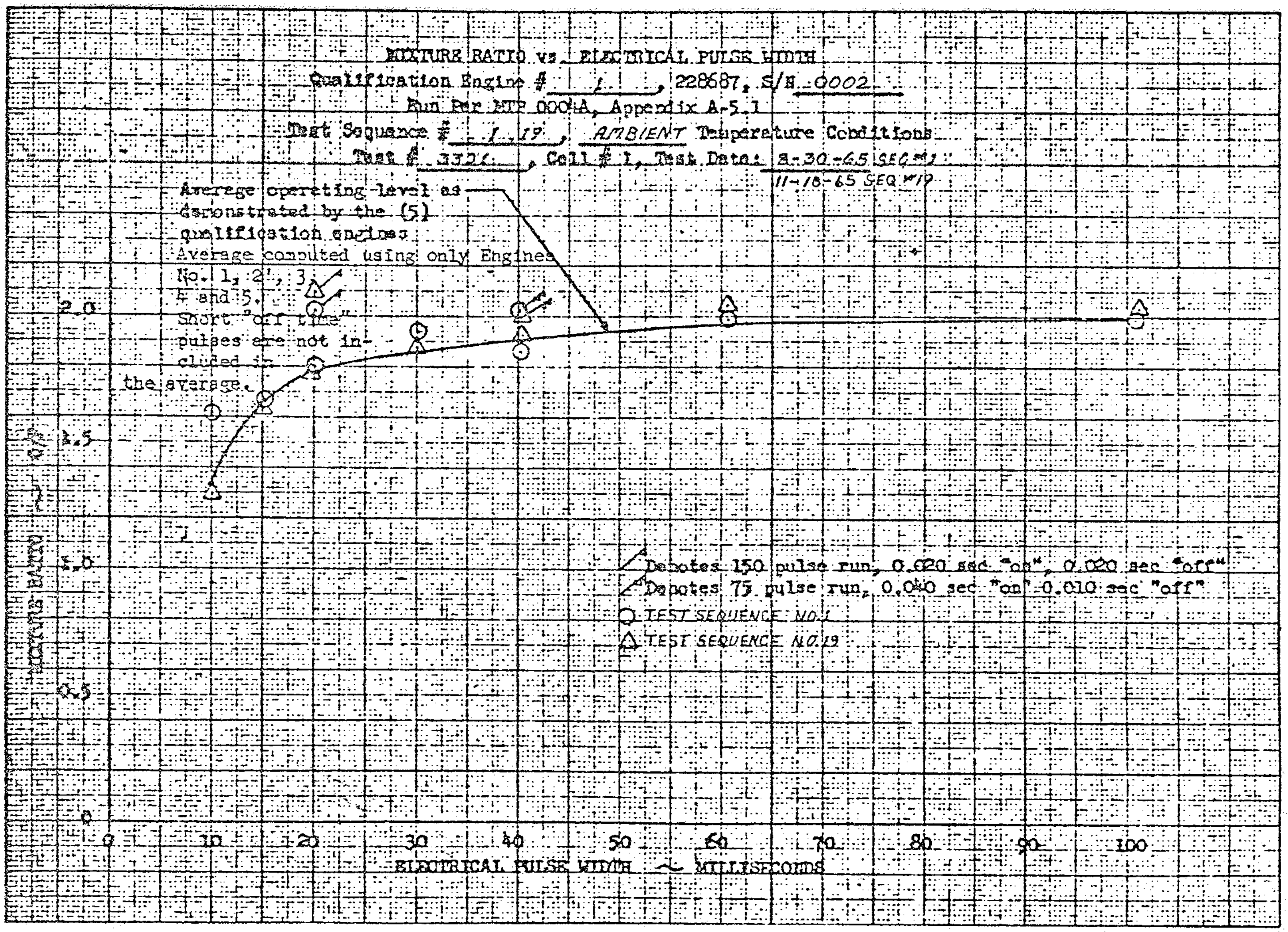
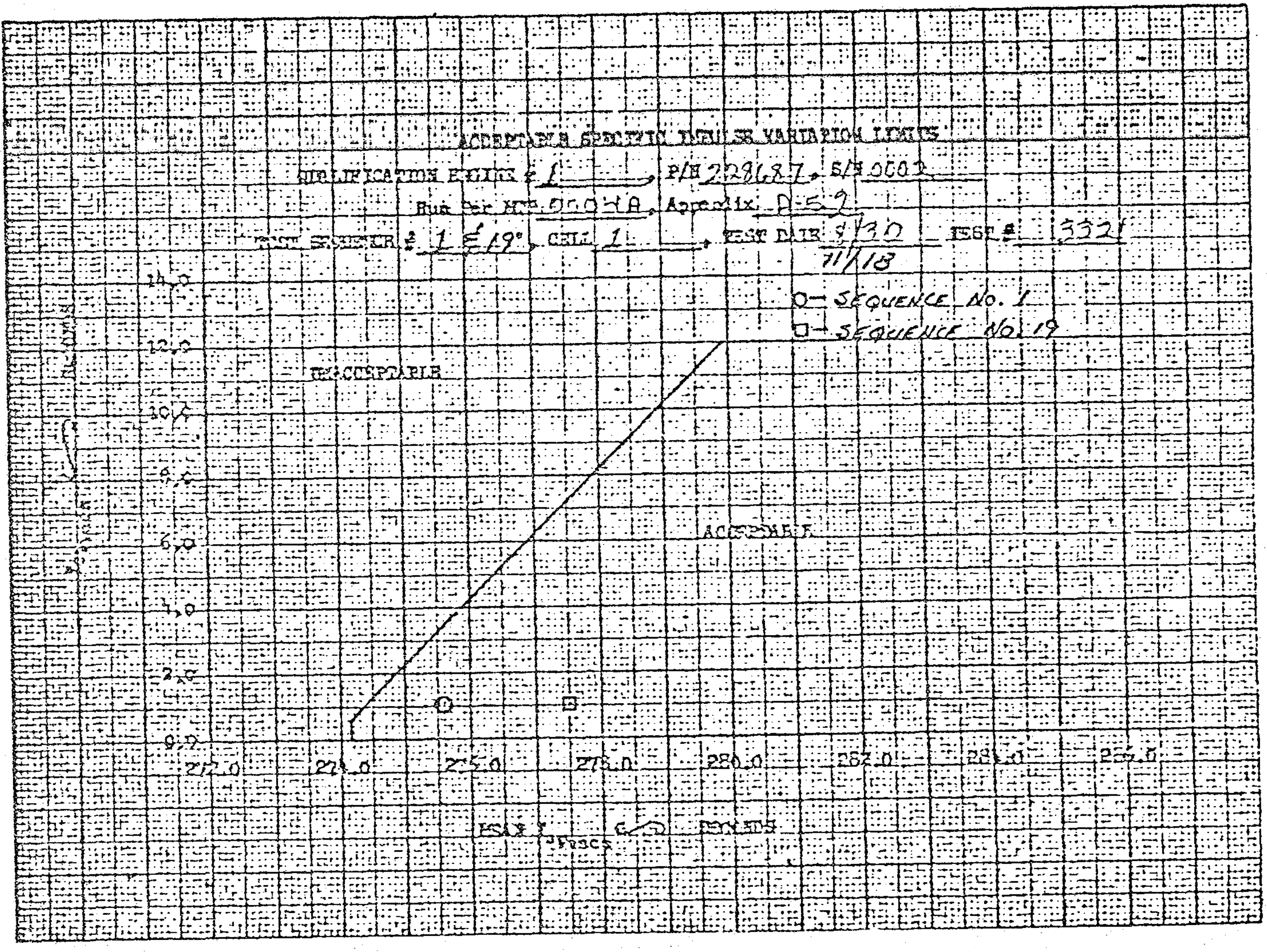


Figure 82

ACCPETABLE SPECIFIC IMPULSE VARIATION LIMITS
QUALIFICATION TEST
ENGINE NUMBER 1



ACCEPTABLE SPECIFIC IMPULSE VARIATION LIMITS

QUALIFICATION ENGINE # 1, P/N 229687, S/N 0002

Run Per ATP 0004A, Appendix D-52

TEST SEQUENCE # 1 & 19, CELL 1, TEST DATE 8/30 71/18 TEST # 3321

Figure 83

ACCEPTABLE THRUST VARIATION LIMITS
QUALIFICATION TEST
ENGINE NUMBER 1

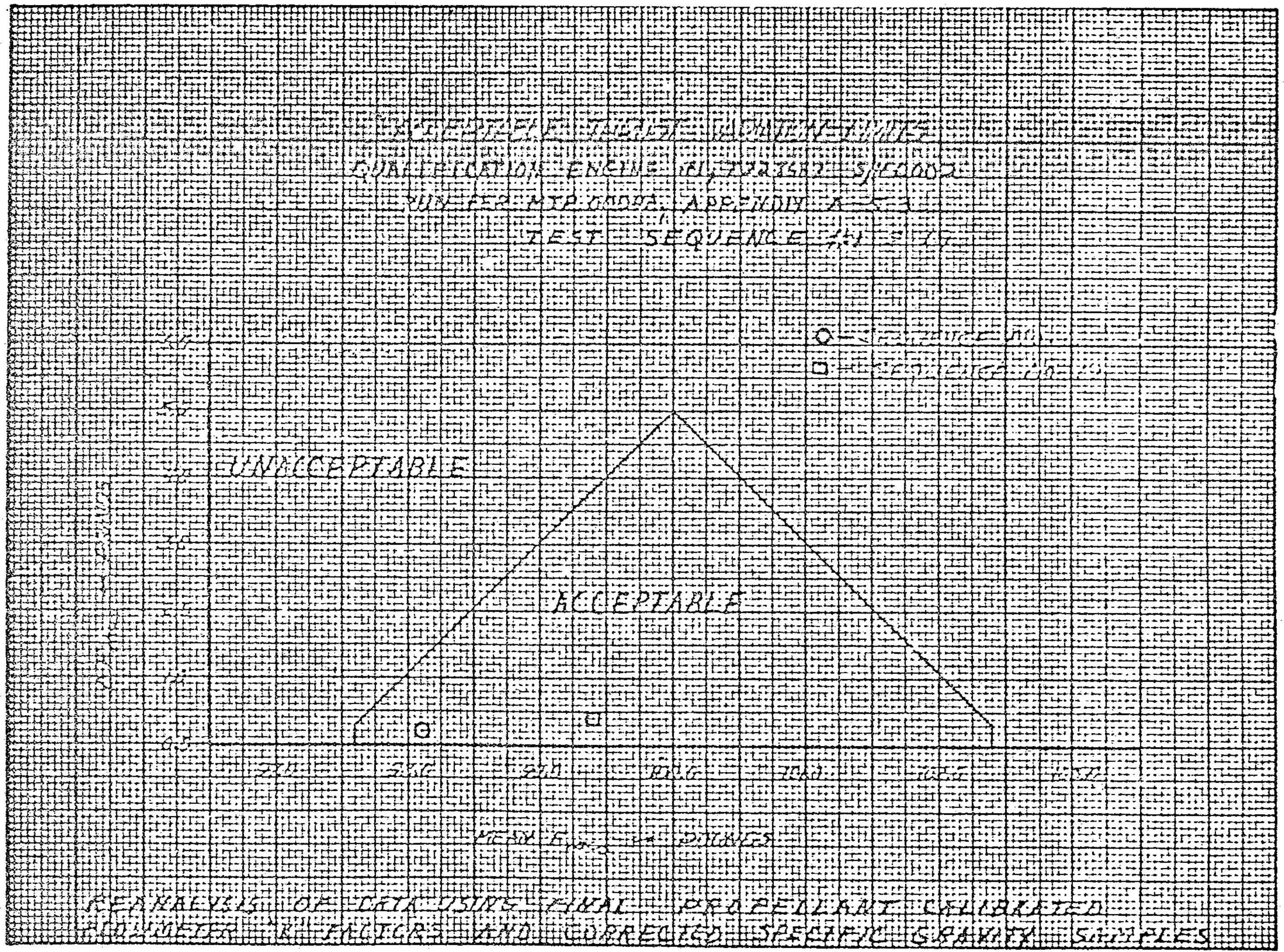
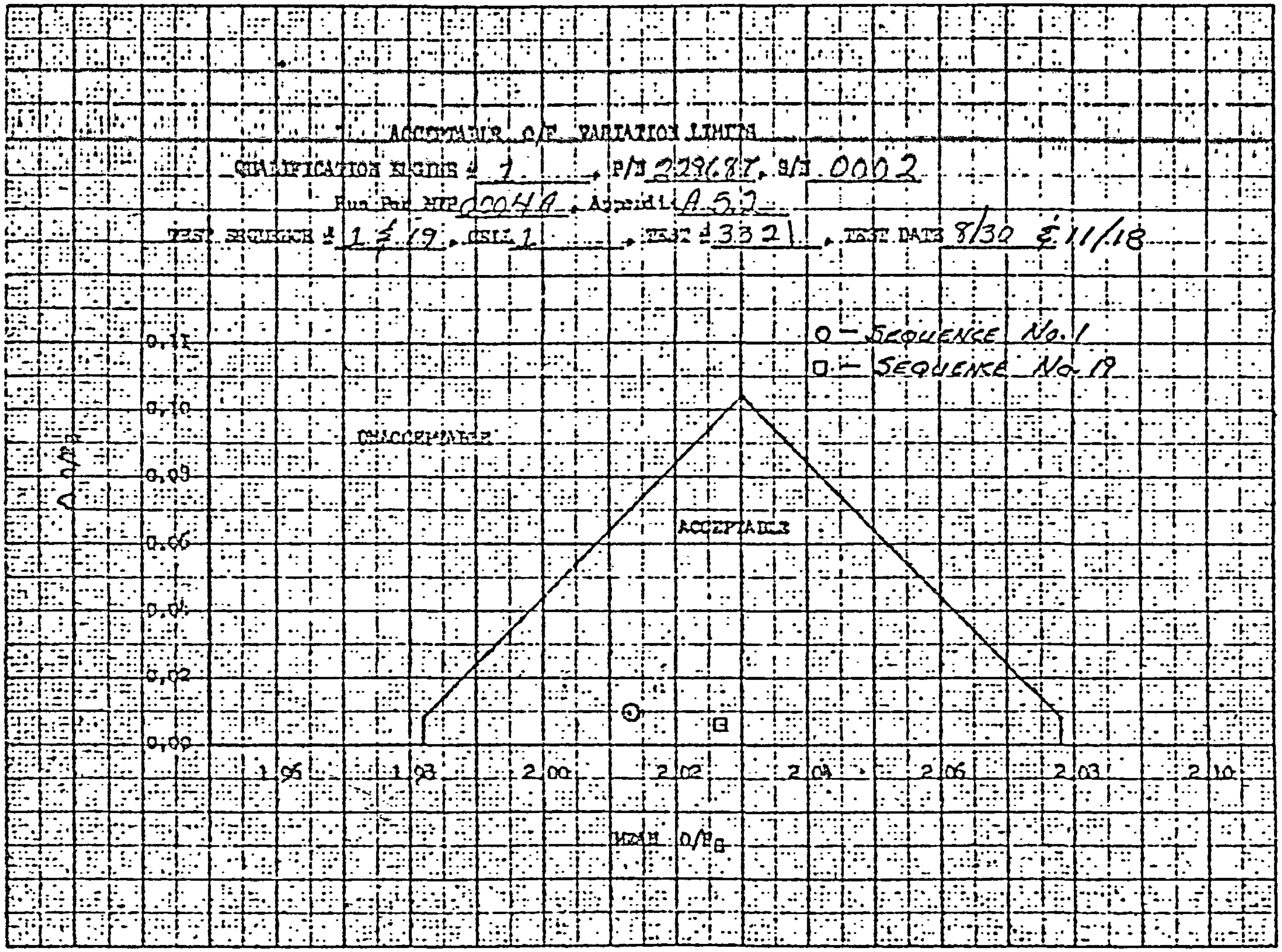


Figure 84

ACCEPTABLE O/F VARIATION LIMITS
QUALIFICATION TEST
ENGINE NUMBER 1



ACCEPTABLE O/F VARIATION LIMITS

QUALIFICATION ENGINE # 1, P/N 229687, S/N 0002

Run Per MTP 0004A, Appendix A 5.2

TEST SEQUENCE # 1 & 19, CELL 1, TEST # 3321, TEST DATE 8/30 & 11/18

Figure 85

At the conclusion of the firing tests, all five engines were capable of further operation, indicating operational capability in excess of that required.

The Qualification Program demonstrated that the performance of the R-4D engine met or exceeded all specification performance requirements that were to be demonstrated, with one relatively minor exception (hot injector thrust decay time for pulses of greater than 40 ms duration). The test provided the first significant statistical sample of data on this engine design. Based on these data, Marquardt recommended specification changes to more accurately define steady state specific impulse, steady state mixture ratio, pulse specific impulse, pulse mixture ratio, and hot injector thrust decay rates.

The program resulted in improved knowledge on engine propellant flow rates. Testing of Engine No. 5 revealed errors in the propellant flowmeter calibrations used both for Acceptance and Qualification testing. These errors were found to be caused by using water instead of propellant as the calibrating fluid. As a result, flowmeters have since been calibrated on propellant.

The program resulted in improved knowledge of the processes that occur during engine shutdown, both in the test cell and in space. Engine No. 2 was damaged during the cold mission simulation test. The resulting failure investigation showed that this situation was due to processes which happen on shutdown and restart in the facility, and would not occur in space. This investigation sheds light on similar instances that have occurred both on this program and on other rocket development programs.

B. Off Limits Test

Immediately following the Qualification Program, an Off Limits Test Program was initiated. The five Qualification Engines were used for these tests and disassembly of the engines was postponed until the completion of the Off Limits Test Program. The purpose of this program was to increase confidence in design margins by conducting tests at off design conditions. Testing was initiated on January 13, 1966 and post checks on all engines were completed by 27 January 1966 except for Engine No. 3, which was completed on 3 March 1966.

A brief description of each test is as follows:

Pulse Operation Survey - Two additional Pulse Operation Surveys were conducted. The pulse survey consisting of subjecting an engine to two surveys of 4000 pulses with a range of pulse widths from 10 to 500 milliseconds and at various repetition rates resulting in 10 to 300 millisecond off times.

Hot Oxidizer Test - One engine was subjected to 60 second engine firings with the oxidizer temperature at 120°F and at 150°F.

2100 Second Run - Two engines were subjected to 2100 seconds of continuous steady state operation.

Boost Vibration - One engine was subjected to increasing vibration to a maximum vibration equal to 3 times the power spectral density listed below:

10 to 90 cps	0.055 g^2/cps at 10 cps with an increase of 3 db per octave to 0.5 g^2/cps at 90 cps.
90 to 250 cps	Constant at 0.5g^2/cps.
250 to 2000 cps	0.5 g^2/cps at 250 cps with a decrease of 3 db per octave to 0.06 g^2/cps at 2000 cps.

Valve Leakage Test - Document engine ignition characteristics when operated with simulated leakage rates of known amounts of oxidizer and fuel. No economical method of simulating controlled leakage was found in the time available. Therefore, at the request of North American, Space Division, further investigations of the optimum controlled leakage method and subsequent testing of Engine No. 5 were terminated.

Post Examination - The engines were subjected to Electrical and Structural Integrity tests. The engines were checked as follows:

The engine was pressure checked to demonstrate structural integrity. The valves were subjected to a transient voltage spike of 50 volts peak with a pulse width of 10 microseconds to demonstrate that they would not be damaged by transient voltage. The valve response and resistance characteristics were measured and compared with characteristics determined during acceptance testing to demonstrate electrical integrity.

The engines were then disassembled for detailed inspection and engineering evaluation.

The Off Limits test matrix and a listing of total valve actuations and burn time for each engine is shown below.

TEST	Qualification Engine Number				
	1	2	3	4	5
Pulse Operation Survey (twice)	×	×			
Hot Oxidizer Test				×	
2100 Second Run	×	×			
Boost Vibration			×		
Post Examination	×	×	×	×	×
Total Valve Actuations	8,062	8,042	29	25	0
Total Burn Time Seconds	3,042	3,062	0	135	0
*Total Valve Actuations	26,553	23,099	17,536	11,488	16,100
*Total Burn Time Seconds	4,532	4,820	1,467	1,528	1,405

*Actuations and burn times include totals accumulated during Qualification Testing and Off-Limits Testing.

Propellants (Controlled Saturation)
Fuel - A-50
Oxidizer - N_2O_4
Pressurant - Helium

All requirements of the Apollo SM RCS Engine Off-Limit Test Program were successfully completed. Engines No. 1 and No. 2 successfully demonstrated the life capabilities of the Part Number (P/N 228687) Engine design. Each engine completed two Pulse Operation Surveys and a continuous 2100 second steady state run. Engines No. 3 and No. 4 documented the design margin of the P/N 228687 engine. Engine No. 3 successfully completed a Vibration Test at three times the level anticipated during the boost phase. Engine No. 4 demonstrated operation with hot propellants up to 150°F. Engine No. 5 was reserved for the valve leak test; however, difficulties in establishing a technique for controlling very low flow leaks precluded testing the engine.

At the conclusion of the above specified tests, the electrical and structural integrity of all four engines was verified by the Post Check Test. The successful completion of this test indicated that all four engines were capable of further operation.

This conclusion was further verified by the Post Examination teardown inspection of the five engines. Inspection of the disassembled parts did not reveal any engine design and/or operational deficiencies which would prevent the engine from exceeding reliable operating and total operating life requirements.

TMC Report A1058 dated 20 May 1966 presented in detail the results of the Off-Limits Test Program.

C. Helium Effects and Ignition Tests

Concurrent with the conduct of the Off-Limits Test Program, a helium effects program was conducted to evaluate the effect of helium saturated propellants on the engine and to conduct ignition tests under saturated propellant conditions. Special equipment was designed, developed and installed in the propellant tanks.

This special saturating equipment consisted of paddles or stirrers which continually agitated the propellant during the saturation period. Samples of the saturated propellant were delivered to Jet Propulsion Laboratory, Pasadena, for analysis. Results showed that propellants, pressurized to 180 psia with mechanical stirrers in the tanks agitating the propellant, would exceed 90% of their ultimate steady state saturation level in one (1) hour of exposure for fuel and four (4) hours for oxidizer. Figures 86 and 87 present these results.

With respect to the ignition tests, important points were uncovered, one being that helium bubbles were being trapped in the propellant system and when one was located properly with respect to the valve injector assembly, large ignition pressures would occur. Secondly, a large ignition pressure experienced at normal zero fuel lead condition was accompanied by long engine ignition delays (greater than 4 ms after last valve full open) and ignition delays of 8 to 9 ms were not uncommon, with 1 to 3 milliseconds being the normal delay. On oxidizer lead starts (6 ms), engine ignition frequently occurred prior to the initiation of fuel flow. This occurred during pulse with 100 ms off time and definitely indicates the presence of residual fuel. Testing continued with further setup modifications including equipment for introducing sized and measured gas bubbles in both the oxidizer and fuel propellant lines plus capability for firing of the engine in the bell up position. Subsequent testing in a vertical up position with predetermined sized bubbles disclosed that no ignition pressure occurred with the engine at temperature conditions of ambient ($T_{head} = T_{throat} = 70 + 20^{o}F$) or cold ($T_{head} = 30 \pm 10^{o}F$; $T_{throat} = 0^{o}F$). The presence of a helium bubble in either propellant systems did, however, affect various ignition parameters. The ignition delay increased as a function of bubble size in both propellant systems, but this effect was more pronounced and orderly with oxidizer bubbles. The nominal mechanical fuel lead was 1.9 ms for these tests; however, the presence of gas

HELIUM-AEROZINE-50 WEIGHT RATIO VS. EXPOSURE TIME

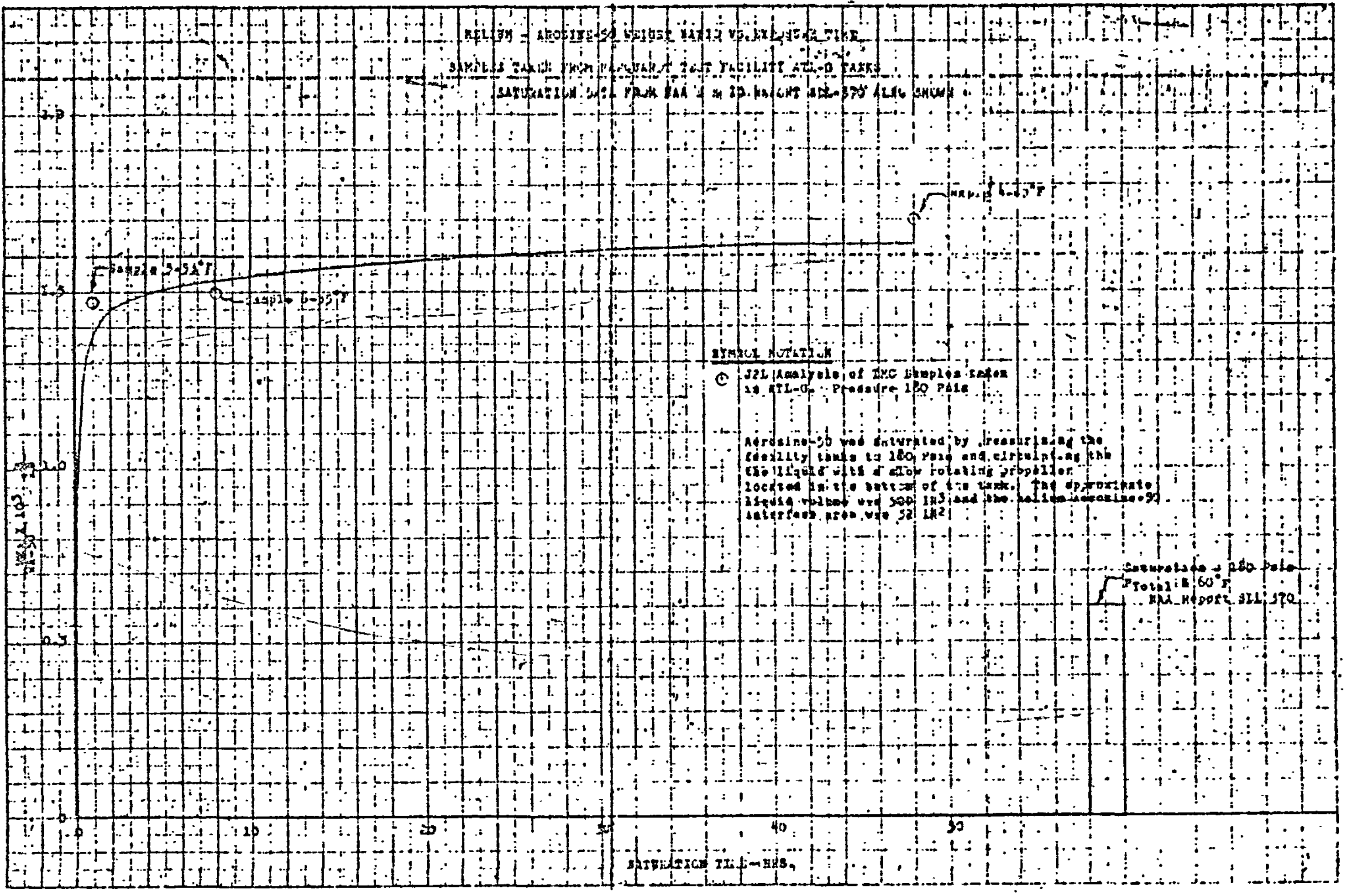


Figure 86

HELIUM - NITROGEN TETROXIDE WEIGHT RATIO VS. EXPOSURE TIME

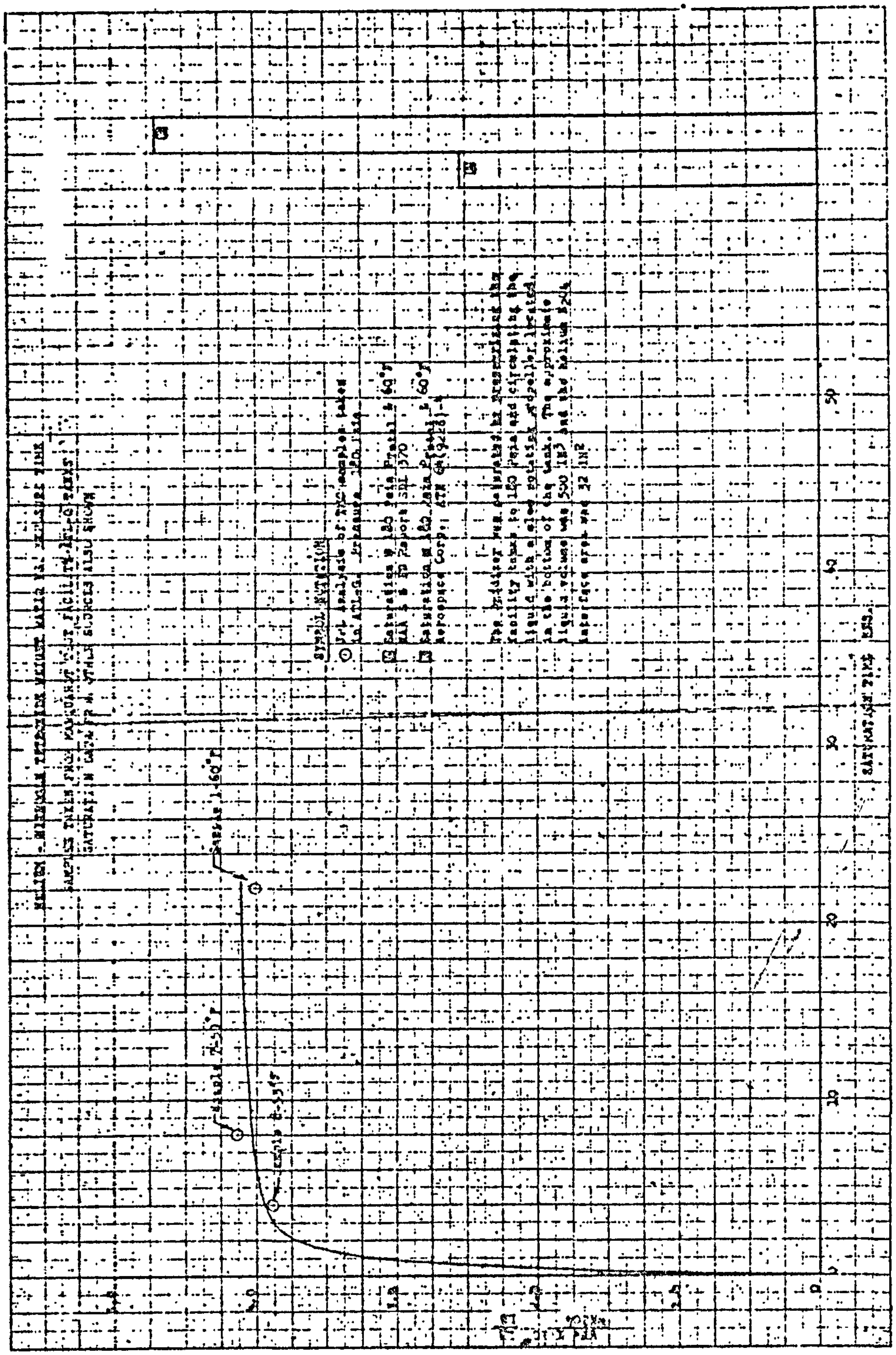


Figure 87

immediately upstream of a valve caused variations in the mechanical lead. The lead varied from 1.2 ms (average) with a .560 in^3 oxidizer bubble to 2.4 ms (average) with a .520 in^3 fuel bubble.

The ignition delay was not influenced by either engine temperature or propellant condition (saturated or degassed) for all tests for which a helium bubble was present. However, for runs without a helium bubble against a valve seat, an influence was observed. The ignition delay for runs with no bubble and degassed propellants ranged from 2.5 ms to 5.5 ms, as opposed to 10 ms with saturated propellants under the same engine temperature conditions. Only one "no bubble" run was made with saturated propellants and a cold engine; the ignition delay for this run was 2.5 ms. Ignition delays of 10 ms were noted for an ambient engine under the same saturated propellant and "no bubble" conditions.

Data obtained during the helium effects program led to a better, although still incomplete understanding of the effects of helium dissolved in the propellants on the SM RCS engine ignition and performance characteristics. In general, the data indicate that helium saturated propellants have an adverse effect on engine performance margins.

Steady state and pulse performance did not appear to be affected by helium saturation of the propellants at design conditions. However, at off design conditions, engine roughness was experienced. Combinations of high head and propellant temperature with high O/F ratios (O/F = 2.2) and/or low thrust levels (90 pounds) resulted in thrust oscillations as large as ± 25 pounds. The limited test results indicate engine performance during periods of thrust oscillation to be lower.

Ignition characteristics of the engine appeared to be compromised during operation with helium saturated propellants. Ignition delay times measured during these tests were significantly longer when the engines were operated on saturated propellants than when operated on unsaturated propellants. High ignition overpressures (with normal fuel lead valve timing) are usually associated with long ignition delays. Results of vertical up firing tests, where known volume helium bubbles were injected immediately upstream of the propellant valves, demonstrated that an effective oxidizer valve lead can occur when helium comes out of the solution into the propellant. Subsequent testing was conducted to determine the regime of engine duty cycles where potentially destructive engine overpressures did not occur. These tests were conducted using oxidizer leads to simulate saturated, bubbly propellant. Test results indicate that engine duty cycle and chamber temperature definitely affect the maximum ignition spike pressure level.

TMC Report S-501 dated 29 June 1966 presents the results of the helium effects program.

The results of the helium effects program pointed out the need to conduct further tests including ignition tests to define the most critical engine operating duty cycle modes at various engine temperatures and to determine the minimum combustor temperature for which safe ignition spike levels can be reliably predicted and controlled with helium saturated propellants.

An arc suppression program was conducted early in 1966 to evaluate the effects of the arc suppression circuit on the performance of the propellant valves. Analysis of test data obtained in testing valves with the arc suppression circuitry in the pulser setup revealed that the direct coil circuit arc suppression affected the valve performance as follows:

1. There was no discernible effect on automatic coil operation.
2. Direct coil opening times are increased by 5 ms for the fuel valve and by 10 ms for the oxidizer valve.
3. Direct coil closing times are increased by a factor of 7 for the fuel valve and by a factor of 9 for the oxidizer valve.
4. During direct coil closing, the oxidizer valve will occasionally close before the fuel valve.

Subsequently, an engine test program utilizing arc suppression circuitry was performed and will be discussed later.

D. Minimum Chamber Temperature Test

As a result of the helium effects program, a minimum chamber temperature mapping program was conducted. The objective was to define "red line" conditions for up attitude firing with saturated propellants at cell pressures less than 0.01 psia.

The major parameters which were specifically evaluated were:

(a) The effect of engine attitude (vertical up or vertical down) on ignition pressure.

(b) A comparison of ignition characteristics for two fuels (MMH and Aerozine-50) when used with N_2O_4 oxidizer.

(c) A comparison of the effects of combustion chamber material (steel, aluminum) on ignition characteristics.

(d) A comparison of ignition characteristics using helium-saturated and unsaturated propellants.

(e) A traverse of engine conditioning temperatures (5°F to 80°F) to determine minimum safe temperature firing range.

(f) Evaluation of firing mode effects, including pulse width, engine "ON" and "OFF" times, programmed ignition lead for oxidizer, cycle frequency, and firing duration, as related to ignition overpressurization.

A total of eight engine tests were run during the program. These tests are summarized in Figure 88 which lists the engine attitude and the propellants employed for each test.

Ignition overpressurization throughout the program can generally be attributed to the "condensed phase" explosion mechanism, where an accumulation of unburnt propellants and combustion products are condensed on the walls of the combustion chamber, as residues from preceding short-duration pulses.

The ignition overpressurization almost invariably occurred on an oxidizer lead pulse, following previous short pulses. The criteria employed for this program for definition of "overpressurization" was a combustion chamber pressure in excess of 750 psia measured on a Kistler pressure transducer.

The level of ignition overpressurization was modified by various influences. Some of these modifying influences, and their results, are as follows:

1. The use of MMH fuel resulted in lower peak ignition pressures than the use of Aerozine-50/N_2O_4 propellant combination.

2. Colder conditioning temperatures for the engine hardware resulted in higher ignition peak pressures.

3. Ignition pressures varied as a function of "OFF" times between pulses. The peak pressures for the Aerozine-50/N204 propellant combination occurred at "OFF" times of 1000 milliseconds, while for the MMH/N_2O_4 combination the highest pressures were at 100 to 200 milliseconds "OFF" times.

4. The combustor material also influenced the level of the ignition overpressurization. This is attributed to the thermal diffusivity of the material.

SUMMARY OF MINIMUM TEMPERATURE IGNITION OVERPRESSURIZATION TESTS

ATL-PAD G

Item	Engine	Date	Test No.	Runs	Attitude	Propellants				Remarks
						Ox	Fuel	Saturated	Unsaturated	
1	12285-001-11	5/8	3407	130-147	Up	Green N_2O_4	MMH	×		Large (≈ 2800) ox manifold pressure
2	11606-001-8	5/12	3407	148-155	Up	Green N_2O_4	MMH		×	Large (> 5000) ox manifold pressure
3	11606-001-9	5/13-14	3407	156-204	Down	Green N_2O_4	MMH		×	Good ignition; largest chamber pressure = 750 psia
4	11606-001-10	5/15, 18	3407	205-240	Down	Green N_2O_4	A-50	×		Movies; good ignition; largest chamber pressure = 1100 psia
5	11606-001-11	5/18-19	3407	241-311	Down	Green N_2O_4	A-50		×	Good ignition; largest chamber pressure = 1850 psia
6	11606-001-12	5/24	3407	312-325	Up	Green N_2O_4	A-50		×	Poor ignition; ox manifold pressure > 5000 psia
7	14100-001	5/26, 6/6	3407	326-774	Down	Green & Brown N_2O_4	A-50	×	×	
8	14200-001	6/13-20	3419	341	Down	Green N_2O_4	A-50 MMH	×		

Figure 88

5. The gas content of the propellant (saturated or unsaturated) had no significant effect on ignition overpressurization for firings which were programmed with oxidizer leads (to simulate a bubble blocking the fuel lead).

6. Long ignition delays were experienced with the engine firing in the "vertical-up" position.

7. The chamber-head seal ring cavity influenced the occurrence and magnitude of the ignition pressure due to an apparent accumulation of residue in the cavity. Evidence of explosions in the cavity was found on this hardware after testing. When this cavity was filled with an O-ring or gasket material, these effects were eliminated.

8. Engine firing attitude apparently affected the frequency of occurrence of both seal ring cavity explosions and injector head oxidizer manifold explosions because of gravity drain of the unburned combustion residues to those areas. Analysis indicates that lower cell pressures would aid in prevention of these problems by increasing residue evaporation rates.

TMC Report A1065 dated 16 November 1966 presents the results of the Minimum Chamber Temperature Evaluation Program.

E. Structural Proof Test

During July 1966, an engine Structural Proof Test Program was initiated. The object of the Minimum Safe Temperature Mapping program using the SM RCS engine with aluminum or steel combustors, Kistler pressure instrumentation, and helium saturated MMH and A-50 fuel, was to determine the minimum temperature where engine damaging ignitions would not occur. Comparison of the measured ignition pressures with molybdenum combustor ultimate fracture pressures, however, was inconclusive because of the lack of knowledge concerning localized stresses in the combustor during a high pressure ignition. The objective of the Proof Test was therefore to demonstrate, by using a molybdenum combustor during testing, what the minimum safe temperature is when using helium saturated A-50 and MMH fuel with "green" (0.4 to 0.8% NO content) NTO oxidizer. The minimum safe temperature with each fuel was to be demonstrated by firing over 500 pulses at numerous duty cycles in the Up, Horizontal and Down firing attitudes without damaging the engine.

Test demonstration of the structural adequacy of the SM RCS engine was accomplished by conducting numerous pulse runs at certain temperature conditions in the Up, Horizontal and Down firing attitudes. The following Table identifies the three engines used, their former designation, and their former usage.

Proof Test Engine	Former Designation	Former Usage
P/N 228686-501, S/N 0002	P/N 228687, S/N 0002	Qual Engine No. 1
P/N 228686-501, S/N 0049	P/N 228687, S/N 0049	Qual Engine No. 2
P/N 228686-501, S/N 0234	P/N 228687, S/N 0234	New Engine

The engines used for these tests were modified by replacing the original chamber expansion bell by a test "bell" which adapts to the test facility duct and collects combustion residue condensed in the bell when firing in the Up attitude. Figure 89 schematically shows an engine with the modified bell in the Up firing attitude as well as instrumentation points. The O-ring flange seals against the facility duct (see Figure 90).

Extensive facility modifications were made for conduct of these tests. These modifications give ATL-Pad G the following capabilities.

1. Low environmental pressure (demonstrated as low as 0.00015 psia).
2. Space radiation simulation using a black surfaced radiation sleeve surrounding the chamber that is maintained at liquid nitrogen temperature.
3. Engine firing in any attitude without facility modification.
4. Video monitoring of preigniter and main chamber combustion.

Figure 90 shows the facility with an engine installed in the Up firing attitude. The very low cell pressure capability is achieved using the Root's blower inline with the steam exhaust system. The large cold trap upstream of the Root's blower is used to condense out unburned propellants (to protect the blower) and to improve pressure recovery time after engine firing. The cold trap requires periodic "defrosting" to eliminate the condensed material that accumulates on it.

ENGINE WITH MODIFIED BELL

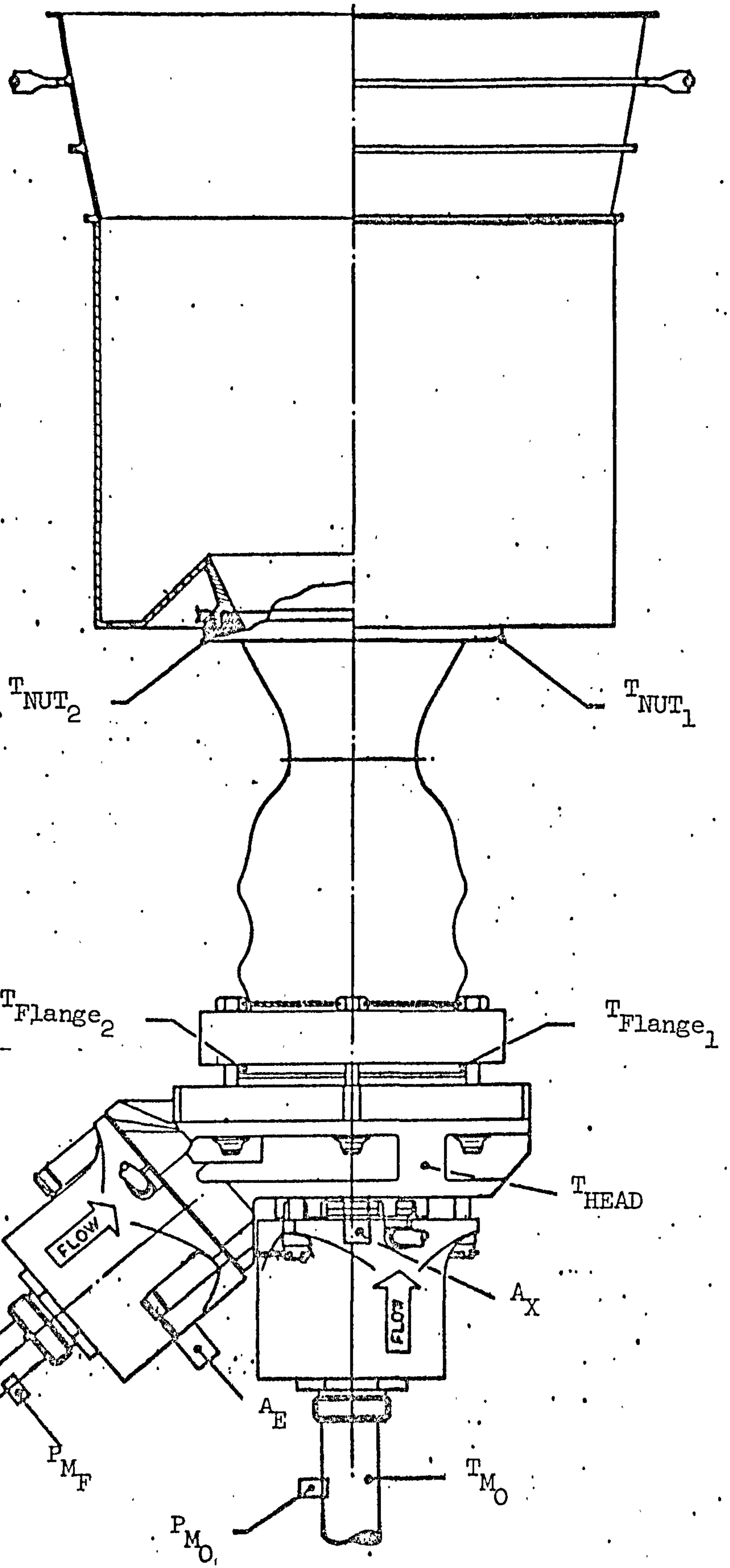
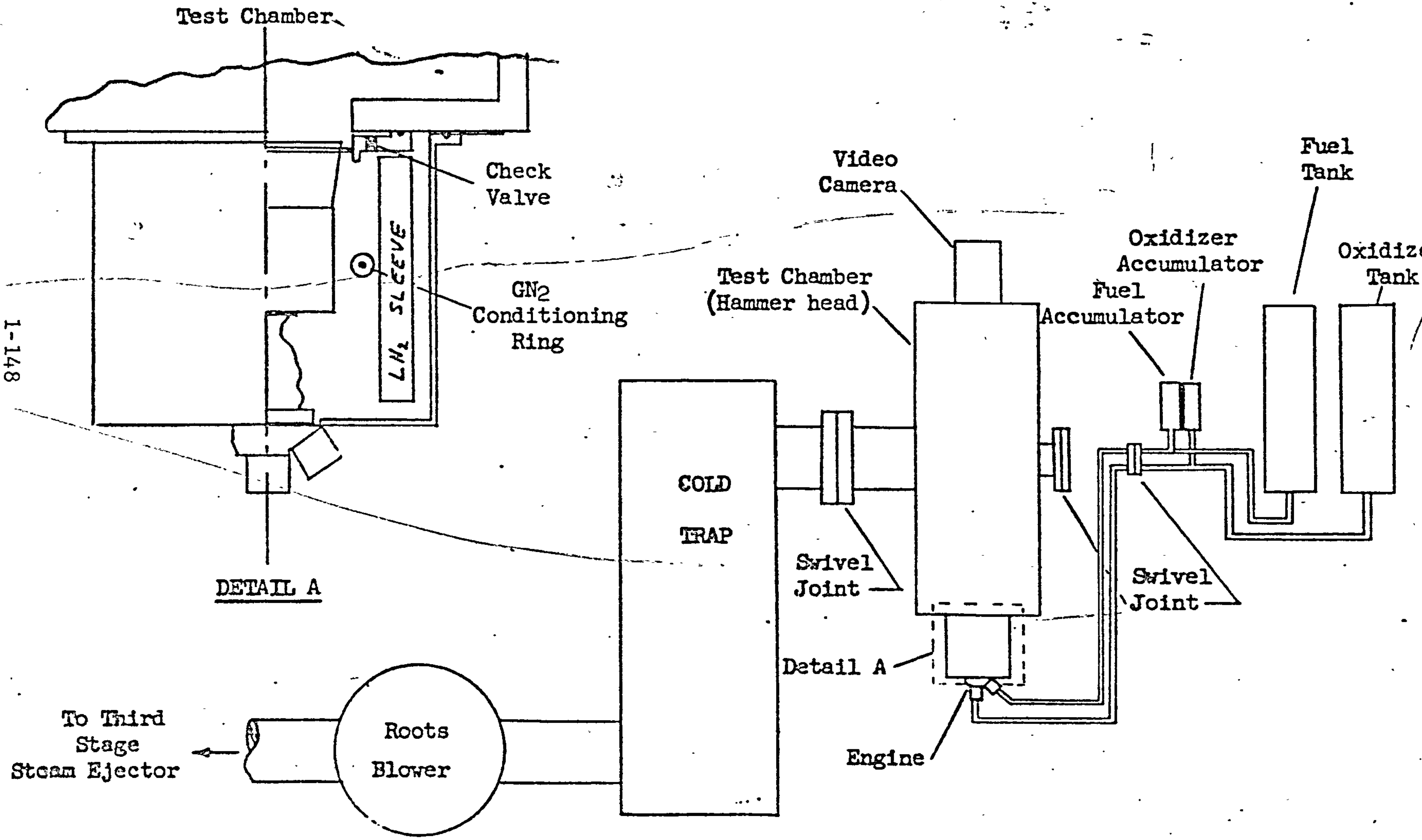


Figure 89

ATL PAD G TEST SETUP SCHEMATIC FOR PROOF TEST



A-1080

The "T" or "hammer head" test chamber with the engine mounted on one end and the TV camera on the other can be rotated, along with the propellant lines, into any desired firing attitude by swiveling at the three locations indicated on Figure 90. The necessity of swiveling the propellant lines made them undesirably long. To preclude long ignition delays that are caused by long propellant lines, gas interface accumulators are installed just upstream of the propellant line swivel. The accumulators are located about six feet away from the engine propellant valves, as compared to about four feet on the SM RCS. The accumulators were intended to function as small propellant tanks.

The environmental can that surrounds the engine consists of a cryogenic sleeve surrounding the engine combustor and bell and a GN_2 ring used for approximate temperature conditioning. To simulate space radiation, the inside of the can was maintained at cell pressure using normally open check valves that close upon engine firing. The check valves prevent combustion products from condensing on the cryogenic sleeve and the engine, thereby affecting the radiation.

The approach used to demonstrate the minimum safe temperature for the SM RCS engine was to test the engine in three attitudes: Up, Horizontal and Down, at various nut temperatures with both A-50 and MMH fuel. Various types of pulse runs were conducted at each condition, and the success or failure criteria was whether or not the engine, particularly the molybdenum combustor, was damaged during this testing. The minimum temperature where a sufficient number and types of runs were made in all three attitudes without combustor failure would be considered the demonstrated minimum safe temperature.

Engine temperature conditions were represented by the bell attach nut temperature. A given nut temperature designated an engine temperature distribution when the nut temperature is steady state in a space cold soak situation; the energy needed to maintain this distribution being supplied by an electrical resistance heater attached to the injector head. The engine component temperature distributions and tolerances as used during the test are given below:

SPACE COLD SOAK ENGINE COMPONENT TEMPERATURES

Nut Temperature	Flange Temperature	Head Temperature	Propellant Temperature
50 ± 5°F	69 ± 2°F	As Required to Obtain T_{nut} and T_{flange}.	40 ± 5°F
40 ± 5°F	58 ± 2°F		40 ± 5°F
30 ± 5°F	47 ± 2°F		40 ± 5°F
20 ± 5°F	36 ± 2°F		40 ± 5°F
10 ± 5°F	26 ± 2°F		40 ± 5°F

The procedure of installing a clean engine after every cold trap defrosting was followed during most of the program so that subsequent testing would be free of residue generated during previous tests. Three engines were used during the test program.

Figures 91 and 92 note the order of engines tested, the dates, attitude, types of runs, etc., for the A-50 and MMH tests, respectively. All pulses were 12 ms in duration. Tests with OFF times of 100, 350, 600, 1000 and 1500 milliseconds were conducted. The last pulse of every run was a programmed oxidizer lead; the other pulses were nominal fuel lead pulses. The last pulse oxidizer leads tested with each engine, each attitude, and each nut temperature are indicated in Figures 91 and 92.

The testing with Aerozine-50 fuel was limited to vertical Up firing on two engines, both of which were fired only at a nut temperature of 50°F. Figure 91 summarizes the A-50 testing. As indicated by Figure 91, molybdenum combustor failure occurred on both engines after more than 300 pulses had been conducted. The S/N 0002 engine combustor failure occurred on a fuel lead pulse, while the S/N 0049 engine combustor failure occurred on a 15 ms oxidizer lead pulse. During the conduct of the vertical Up testing with A-50, it was observed on the TV monitor that condensed combustion residue accumulated in the bell collector ring as testing progressed. Testing with A-50 as the fuel was discontinued when a joint NR/SD-TMC decision was made to abandon the objective of demonstrating a safe engine temperature condition with A-50 and to proceed with MMH fuel.

Testing was accomplished in the Up, Horizontal and Down firing attitudes with MMH fuel. In the Up firing attitude, tests were conducted at nut temperatures ranging from 20° to 50°F; while in the Horizontal attitude, the nut temperature range was 10° to 50°F. Tests were conducted only with a 30°F nut temperature in the Down firing attitude. A summary of the MMH testing is given in Figure 92.

Initially, tests with MMH were conducted in the Up firing attitude with nut temperatures of 50°F, 40°F, 30°F and 20°F without combustor failure. Following tests in the Horizontal attitude where 30°F was determined as the minimum safe nut temperature, additional runs were conducted in the Up attitude with a 30°F nut temperature. A total of 1,041 pulses at numerous duty cycles were conducted in the Up attitude with a 30°F nut temperature.

In the horizontal firing attitude, runs were conducted at 50°F, 40°F, 30°F and 20°F nut temperature conditions without engine damage. A combustor failed at 10°F, and it was decided that 20°F was probably marginal. Thus, 30°F was decided to be the minimum safe temperature in the horizontal attitude. Additional pulses were repeated at 30°F, bringing the total at this condition to 615.

SUMMARY OF PROOF TEST PROGRAM (A-50 FUEL)

Engine	Attitude	T_{nut} °F	Total Runs	Total Pulses	OFF Times (ms)	Ox Leads (ms)	No. of Pulses per Run	Date	Comments
228686-501 S/N 0049	Up	50	58	309	350, 600 1000, 1500	6, 8, 15	4, 7	8-4-66	Combustor failed on 309th pulse (15 ms ox lead)
228686-501 S/N 0002	Up	50	68	347	350, 600 1000, 1500	2, 6, 8	4, 7	8-6-66	Combustor failed on 345th pulse (nominal fuel lead)
		Totals	126	656					

Figure 91

Engine	Attitude	T_{nut} °F	Total Runs	Total Pulses	OFF Times (ms)	Ox Leads (ms)	No. of Pulses per Run	Date	Comments
228686-501 S/N 0234	Horizontal	40	22	142	100, 350, 600	6	4, 7, 9	8-14-66	No engine damage
228686-501 S/N 0002	Horizontal	50	45	303	100, 350, 600, 1000 1500	6, 8, 15	4, 7, 9	8-16-66	No engine damage
228686-501 S/N 0002	Horizontal	40	45	300	100, 350, 600, 1000 1500	6, 8, 15	4, 7, 9	8-17-66	No engine damage
228686-501 S/N 0234	Horizontal	30	90	607	100, 350, 600, 1000 1500	6, 8, 15	4, 7, 9	8-19-66	No engine damage
228686-501 S/N 0049	Down	30	90	604	100, 350, 600, 1000 1500	6, 8, 15	4, 7, 9	8-19-66	No engine damage
228686-501 S/N 0002	Up	30	54	555	100, 350 600, 1000 1500, 5000 & 15,000	0, 2, 6, 8, 15	4, 7, 9	8-24-66	No engine damage
		Totals =	749	5225					

Figure 92

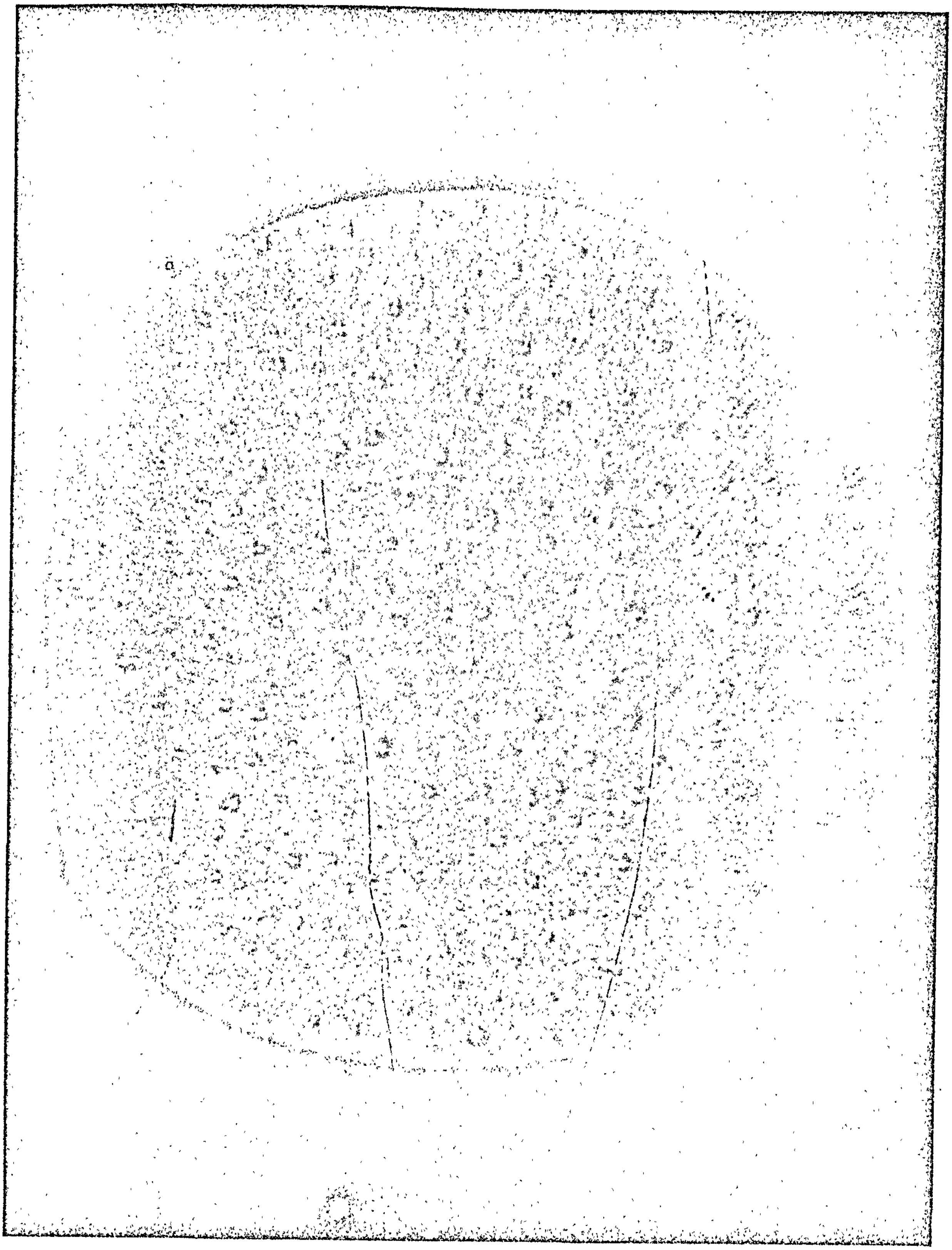
Only a 30°F nut temperature was tested in the Down firing attitude. No engine damage was incurred during these tests. No nut temperature other than 30°F, which was established as the minimum safe temperature in the Horizontal firing, was tested due to the abundance of previous TMC ignition test data in the Down firing attitude. A total of 604 pulses were fired at this condition.

Figure 92 indicates a variation in the numbers of pulses, number of runs, last pulse oxidizer lead, etc., were conducted at each nut temperature and attitude. The number of pulses and variety of types of runs conducted at a 30°F nut temperature in all three attitudes is considered adequate to demonstrate safe ignition at this temperature. No combustor failure occurred with a 20°F nut temperature, but fewer runs were conducted than at 30°F, thus, less confidence existed as to the safety of firing at this condition.

During the conduct of the Up firing tests with MMH, combustion residue was observed accumulating in the bell collector ring as the testing progressed. After the engines were removed from the facility, samples of this residue were taken and analyzed. It was also observed that the inside of the combustors appeared clean shortly after engine removal from the test facility; but after several minutes exposure to ambient air, small liquid beads appeared on the combustor wall. These beads grew in size and number with time and were fairly evenly distributed. Samples, as well as photographs, were taken of this residue.

Samples of combustion residue found in the engine combustor and bell collector ring were analyzed to determine their composition. Analysis of the samples taken from both A-50 and MMH tests showed that hydrazine ions and nitrate ions existed in all samples with occasional determination of the presence of ammonium ions. The amount of residue found in the bell collector ring after tests with MMH was somewhat less than the residue found after A-50 tests. The equipment used to analyze this residue was inadequate to further define the components; e.g., hydrazine nitrate, ammonium nitrate, etc. These compounds can only be speculated to exist. Also, the technique used to identify the components present in this residue was qualitative only, and was unable to determine the amounts or quantity of each component present.

Immediately after each engine's removal from the facility, a visual inspection of the inside of the molybdenum combustor showed little noticeable residue to be present. Small beads of liquid began appearing in great numbers on the wall after about 15 to 30 minutes exposure to ambient air. Photographs of these beads were taken, with Figure 93 being a representative example. It seems that a small layer of crystals remains on the combustor wall after engine firing. This crystal layer cannot be easily seen until it absorbs moisture and becomes a liquid, which happens when it is in contact with ambient air. Samples of this residue proved to contain the same ions as the samples taken from the bell collector ring.



NFG. T3440-39

Residue on Inside of Combustor 20 Minutes after Removal from Pad G - Engine P/N 228686-501, S/N 0002

Figure 93

Laboratory tests were also conducted to demonstrate that unreacted fuels, as well as some of the propellant reaction products, are explosive under certain conditions. Samples of hydrazine, A-50, MMH and crystals formed in the reaction between MMH and dilute nitric acid were heated slowly in confined metal tubes. In each case, the tube was burst with explosive violence when a sufficiently high temperature was reached. The crystals formed in the reaction between MMH and dilute nitric acid were believed to be the same as the residue found in the engines after firing with MMH.

The following conclusions were reached from this program:

1. An engine space temperature condition corresponding to a 30°F bell attach nut temperature is the minimum condition where the SM RCS engine can be safely ignited in all attitudes with MMH fuel and "green" NTO oxidizer.

2. The minimum safe space temperature condition for the SM RCS engine with A-50 fuel is greater than that corresponding to a 50°F bell attach nut.

3. Combustion residue is left in the engine when firing with both MMH and A-50 fuel. The residues left after firing with both fuels, as well as samples of the fuels themselves, explode when they are heated in a confined volume.

TMC Report A1066 dated November 11, 1966 presents the detailed report on this Structural Proof Test Program.

F. Lunar Module Design Verification Ignition Test

After the Service Module Structural Proof Program, a Lunar Module Design Verification Engine Ignition Test Program was conducted. The purpose of this program was to determine the minimum engine flange temperature at which safe ignition would occur when the Lunar Module (LM) RCS engines (same as used on Service Module) were fired vertically Up, Horizontal and Vertical Down attitude with helium saturated propellants, Aerozine-50 fuel and "green" nitrogen tetroxide oxidizer. This program was conducted in a manner quite similar to the Service Module Structural Proof program, the same engines were used in order to save costs.

It was concluded that safe engine ignition did occur in this test program with saturated Aerozine-50 fuel regardless of attitude at a flange temperature of 80°F or greater.

TMC Report L1038 dated 23 November 1966 presents the results of this program.

G. SM RCS Supplemental Qualification Test

After the Structural Proof Test Program, the fuel for the SM RCS was changed from Aerozine-50 to MMH because the ignition characteristics of MMH were better. Accordingly, a Supplemental Qualification Test program was conducted with MMH. The objective of the test was to qualify the Service Module Reaction Control Engine as an Apollo manned flight item when utilizing helium saturated monomethylhydrazine and nitrogen tetroxide, with a nitric oxide content of 0.4 to 0.8% by weight, as the propellants; and with an engine electrical command system that incorporates arc suppression circuitry designed to limit the amount of overvoltage induced when the engine is operated on direct (manual) coils.

The test was conducted per Marquardt Test Plan (MTP) 0056. Testing started on October 8, 1966 and was completed on November 9, 1966. The test sequences to which the three engines were subjected is presented in Figure 94. These were similar to the tests in the Qualification tests and a description of them can be found in Section V-A.

Engines No. 1 and 3 were used in the SM RCS Qualification program and Structural Adequacy Proof Test, respectively; and to reduce costs, were refurbished for use in the Supplemental Qualification program. This refurbishment included new combustors, valve seats, valve armatures, valve springs and seals. Engine No. 2 was randomly selected and purchased from the common engine production line. Prior to the Supplemental Qualification Test, each of the three engines was required to undergo and pass an Ambient Calibration Test with nonsaturated Aerozine-50 as the fuel and nonsaturated nitrogen tetroxide as the oxidizer. Tabulated on Figure 95 are the starts and burn time accumulated by the three engines during the test.

Figures 96 through 99 show the steady state and pulse performance for the Number 1 engine.

The three engines successfully completed the required tests, demonstrating that the Apollo R-4D SM RCS engine will operate safely under the environmental conditions for which it was designed when utilizing helium saturated monomethylhydrazine as the fuel and helium saturated nitrogen tetroxide as the oxidizer.

TMC Report A1068 dated 7 December 1966 presents the detailed results of the R-4D Supplemental Qualification Test.

SUPPLEMENTAL QUALIFICATION TEST MATRIX

Test Sequence No.	Test	Engine No.		
		1	2	3
1	Calibration - Ambient (Nonsaturated propellants)	X	X	X
2	Calibration - Hot (Nonsaturated propellants)	X		X
3	Calibration - Cold (Nonsaturated propellants)	X	X	
4	Calibration - Ambient	X		X
5	Mission Simulation (Cold) Part 1	X		
6	Mission Simulation (Cold) Part 2		X	
7	Mission Simulation (Cold) Part 3	X		
8	Mission Simulation (Cold) Part 4		X	
9	Mission Simulation (Ambient)			X
10	Mission Simulation (Hot)			X
11	Pulse Temperature Survey			
12	Direct Coil Duty Cycle			X
13	Orbit Retrograde			X
14	Calibration - Ambient (Nonsaturated propellants)	X	X	X
15	Electrical and Structural Integrity	X	X	X

Figure 94

R-4D SUPPLEMENTAL QUALIFICATION
ENGINE BURN TIME AND VALVE CYCLE TABLE

ENGINE NO.	ACTUAL BURN TIME (sec.)	RELIABLE BURN TIME (sec.)	ACTUAL NO. OF PULSES	RELIABLE NO. OF PULSES
1	492.367	282.95	8,225	4,575
2	290.346	85.89	5,977	2,066
3	1320.69	1112.04	13,175	8,031

Note:

The reliable burn time and reliable number of pulses refers to the total actual burn time less those burn seconds and number of pulses accrued during the ambient calibration test with A-50, the first ambient calibration test with MMH, and the final ambient calibration test with MMH. This definition of reliable burn time and reliable number of pulses applies to the Supplemental Qualification Program only by mutual agreement between TMC and NAA/S&ID.

Figure 95

THE Marquardt CORPORATION
VAN NUYS, CALIFORNIA

SUPPLEMENTAL QUALIFICATION TEST
STEADY STATE TEST DATA (RECORD RUNS)

Non-saturated propellants(Fuel- MMH)(Ox- N_2O_4 GREEN)

ENGINE ASSEMBLY 228686-501 SUPPLEMENTAL QUAL ENGINE # 1 S/N 0049 PAGE 1 OF 1

TEST NO. 3438 CELL NO. 1 TEST DATE 10-9-66

M.T.P. 0056 APPENDIX PARAGRAPH A-4.3 SEQUENCE # 1

INPUT	RUN NO.	TIME	$\dot{w}_o$ - cps	T_{fm_o} - °F	$\dot{w}_f$ - cps	T_{fm_f} - °F	F_{test} - lbs	P_{CELL} - PSIA	$P_{c_{TEST}}$ - PSIA	P_{m_o} - PSIA SET	P_{m_o} - PSIA INLET	P_{m_f} - PSIA SET	P_{m_f} - PSIA INLET
OUTPUT	SG_o	SG_f	$\dot{w}_{o_{TEST}}$ - pps	$\dot{w}_{f_{TEST}}$ - pps	$\dot{w}_{P_{TEST}}$ - pps	O/F_{TEST}	$F_{vac_{TEST}}$ - lbs	$I_{sp_{vac_{TEST}}}$ - sec	$C_{f_{vac_{TEST}}}$	C^*_{TEST} ft-sec	ΔP_o - psi	ΔP_f - psi	
	3951	5	1302.	81.7	1117.	81.0	98.0	.0692	95.5	183.5	169.6	175.9	169.4
	1.4288	.8683	.2413	.1167	.3580	2.067	99.6	278.3	1.764	5075.	74.1	73.9	
	3952	5	1295.	72.8	1112.	73.6	97.9	.0692	94.9	183.2	169.4	175.7	169.3
	1.4403	.8721	.2419	.1167	.3586	2.073	99.5	277.5	1.772	5038.	74.5	74.4	
	3953	5	1294.	74.7	1111.	75.1	98.0	.0692	95.3	183.2	169.4	175.7	169.2
	1.4378	.8714	.2413	.1165	.3578	2.071	99.6	278.5	1.768	5067.	74.1	73.9	
	3954	5	1293.	73.6	1113.	73.5	97.9	.0702	95.5	183.3	169.5	175.8	169.3
	1.4392	.8722	.2413	.1168	.3581	2.066	99.5	277.9	1.762	5074.	74.0	73.8	

MEAN Fvac	Δ Fvac	MEAN Isp	Δ Isp
99.6	0.1	278.1	1.0

Figure 96

SPECIFIC IMPULSE VS. ELECTRICAL PULSE WIDTH
SM SUPPLEMENTAL QUALIFICATION TEST
ENGINE NUMBER 1

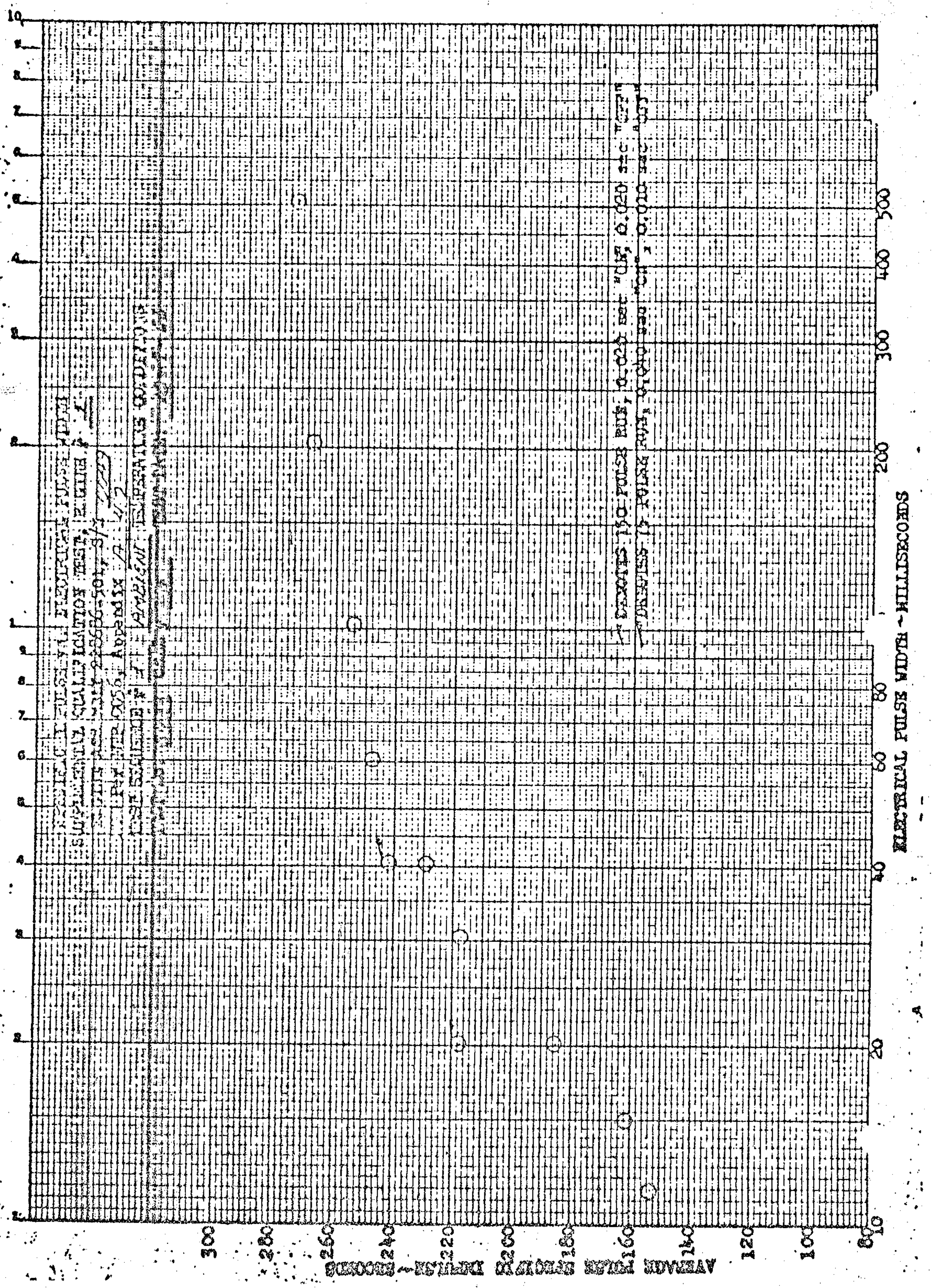


Figure 97

TOTAL IMPULSE VS. ELECTRICAL PULSE WIDTH
SM SUPPLEMENTAL QUALIFICATION TEST
ENGINE NUMBER 1

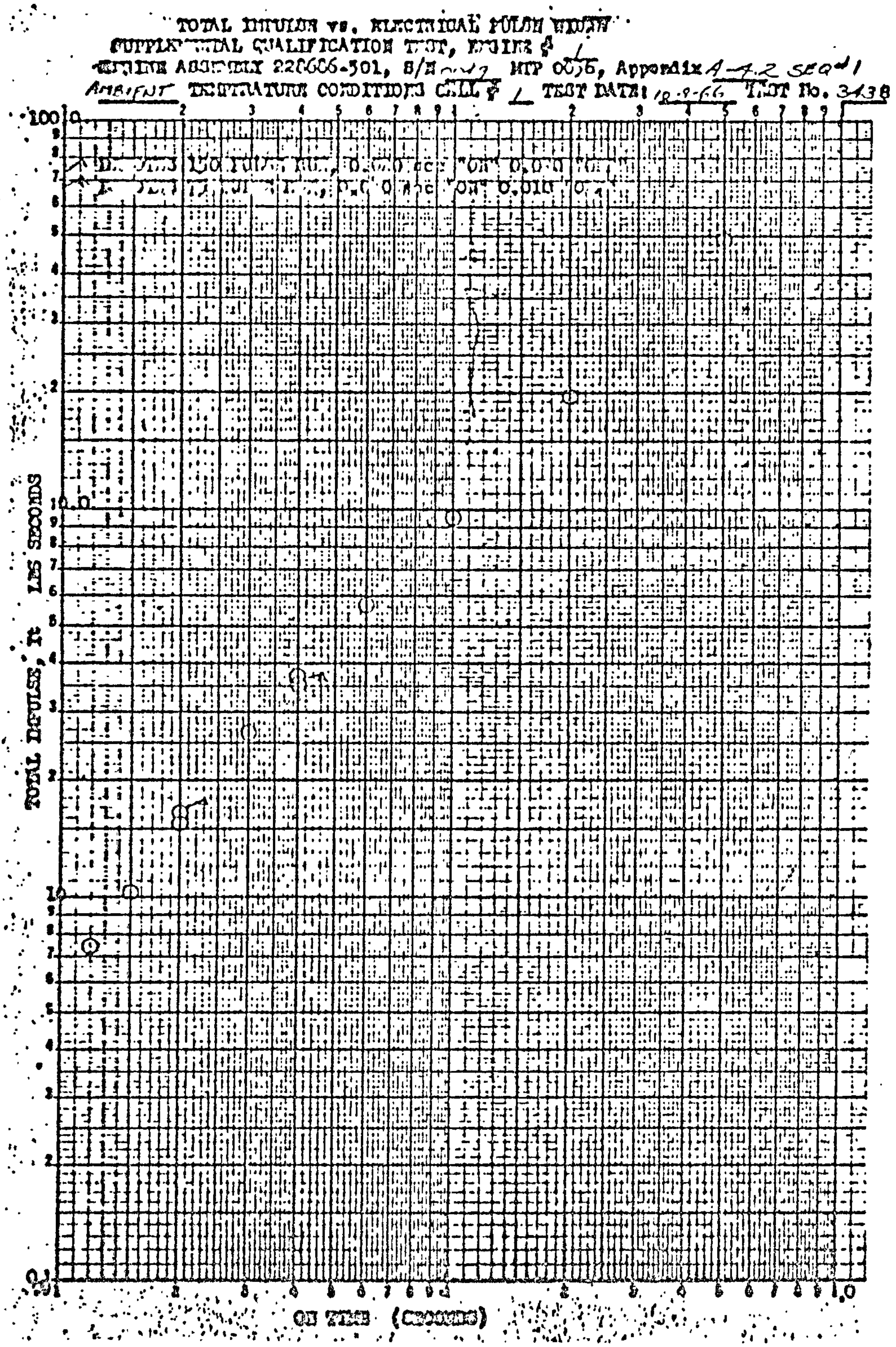


Figure 98

MIXTURE RATIO VS. ELECTRICAL PULSE WIDTH
SM SUPPLEMENTAL QUALIFICATION TEST
ENGINE NUMBER 1

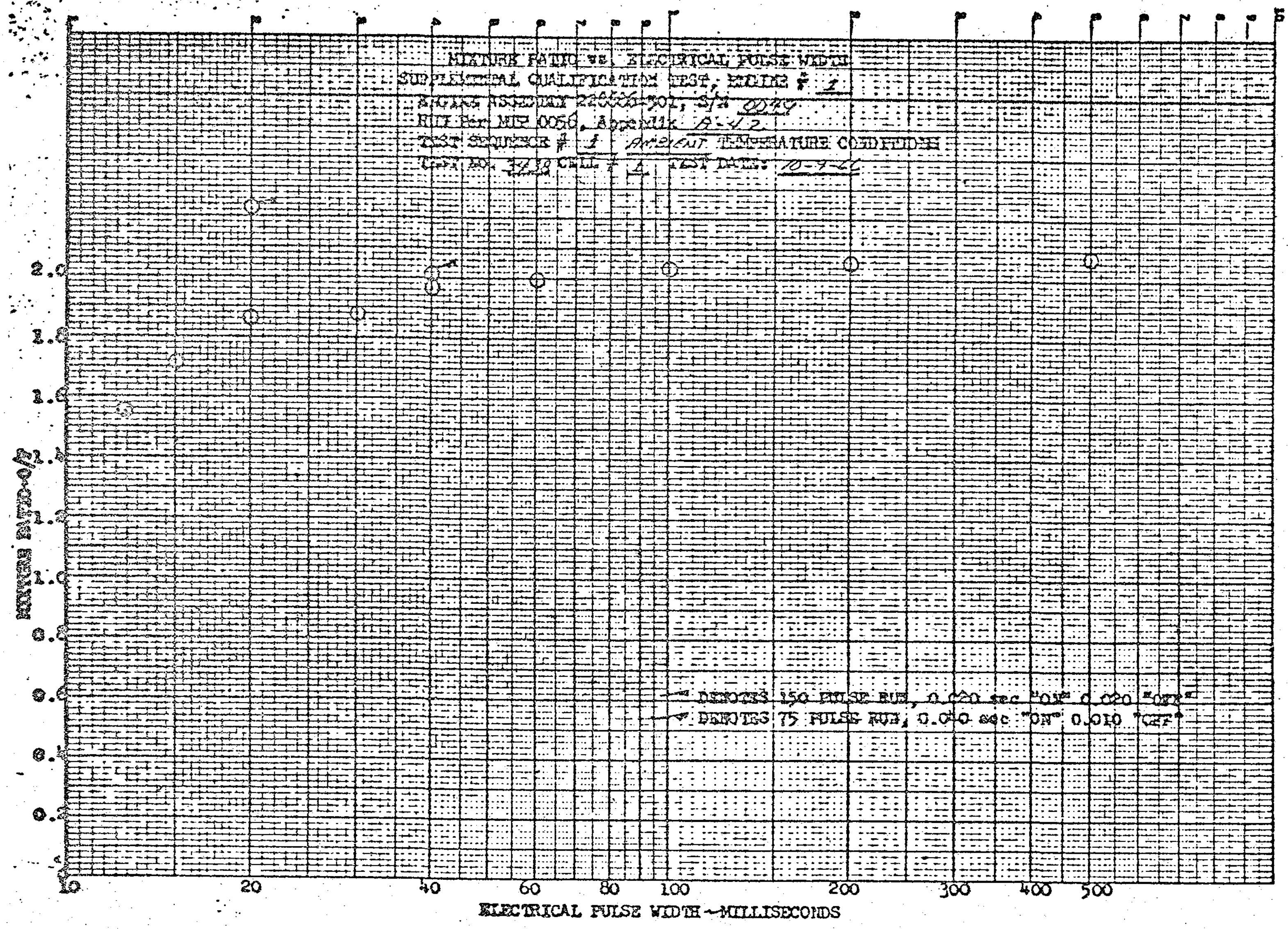


Figure 99

H. LM RCS Supplemental Qualification Test

After the Lunar Module Design Verification Ignition Test, heaters were added to the RCS engines to hold the flange temperature at 120°F, since the test had shown that safe ignition will occur with saturated Aerozine-50 regardless of engine attitude at a flange temperature of 80°F or greater. Accordingly, a Supplemental Qualification test program was conducted with Aerozine-50 as the fuel and with the flange temperature controlled to 120°F. The objective of the test was to qualify the Lunar Module Reaction Control Engine assembly as an Apollo manned flight item when utilizing helium saturated Aerozine-50 and nitrogen tetroxide, with nitric oxide content of 0.4 to 0.8% by weight, as the propellants; with an engine electrical command system that incorporates arc suppression circuitry designed to limit the amount of over-voltage induced when the engine is operated on direct (manual) coils; and with flange temperature controlled to 120°F.

The test was conducted per Marquardt Test Plan (MTP) 0059. Testing started on 23 November 1966 and was completed on 20 January 1967. The test sequences to which the two engines were subjected is presented in Figure 100. A description of each test is given below.

Calibration Tests - Appendix A of MTP 0059

Four types of calibration test were conducted; ambient, hot, and cold temperature condition tests, all with saturated propellants, and one ambient calibration test (Referee run) with nonsaturated propellants. In all, ten calibration tests were conducted, each engine being subjected to five calibration tests. These tests were designed to serve as an index and monitor of engine performance at the above stated conditions. Each test consisted of four 5-second steady state runs and eleven pulse mode runs with burn times ranging from 0.013 to 0.500 seconds.

Mission Simulation (Cold) - Appendix B of MTP 0059

These tests were conducted in two parts. Part A consisted of subjecting the engines to a series of taped duty cycles which were generated by the Apollo Mission Simulator. All Part A tests used automatic coils, and saturated propellants. Part B of the Mission Simulation Test (Cold) was designed to demonstrate safe engine operation when commanded by the manual (direct) valve coils. All direct coil operation was conducted from a pre-programmed pulser.

All Mission Simulation Tests were pre-qualification type tests. Two types of chamber pressure instrumentation were used; flight type chamber pressure transducers, (TMC P/N 228658) and flight type chamber pressure switch (GAEC LSC 310-651). Two engine attitudes were tested; horizontal and vertical up.

SUPPLEMENTAL QUALIFICATION TEST MATRIX

TEST SEQUENCE NO.	TEST NAME	ENGINE NO.	
		1	2
REF	REFEREE RUN	X	X
1	CALIBRATION - AMBIENT	X	X
2	CALIBRATION - HOT	X	X
3	CALIBRATION - COLD	X	X
4	MISSION SIMULATION - COLD	X	
5	MISSION SIMULATION - COLD		X
6	MISSION SIMULATION - COLD	X	
7	MISSION SIMULATION - COLD		X
8	MISSION SIMULATION - COLD	X	
9	MISSION SIMULATION - COLD		X
10	MISSION ABORT		X
11	CALIBRATION - AMBIENT	X	X
12	ELECTRICAL AND STRUCTURAL INTEGRITY	X	X

Figure 100

Mission Abort - Appendix C of MTP 0059

This test, conducted on one engine, consisted of intermittent steady state runs with an accumulated engine firing time of 350 seconds followed by 150 pulses from a pre-programmed pulser.

Electrical - Structural Integrity Test - Appendix D of MTP 0059

This test consisted of a series of electrical and mechanical bench tests designed to demonstrate the post burn electrical and structural integrity of the engine. This was the final test to be conducted on each participating engine.

Ambient Referee Run (Non-Saturated Propellants)

The Referee Run Test consisted of firing the engine in both steady state and pulse modes with firing durations ranging from 0.013 seconds to 5.00 seconds. The Referee Run was the first burn test to be conducted on each participating engine.

The two engines used during the test were randomly selected and purchased from the Common Engine production line. The starts and burn time as accumulated during the test are tabulated in Figure 101.

Both engines successfully completed the required tests, demonstrating that the LM RCS engine will operate safely with flange temperatures controlled to 120°F when utilizing helium saturated Aerozine-50 as the fuel and helium saturated nitrogen tetroxide as the oxidizer.

TMC Report L-1041 presents the detailed results of the LM RCS Supplemental Qualification Test.

Engine No.	Actual Burn Time (Sec)	Actual Number of Pulses
1	437.907*	7873*
2	838.673*	Valve #1 4246* Valve #2 5737

*Includes Referee Run Actuation and Burn Time as follows:

Engine No. 1, 1262 actuations - 68.965 secs;
Engine No. 2, 1178 actuations - 76.550 secs.

L/M Supplemental Qualification Engine Burn Time and Valve Cycle Table

Figure 101

VI. CONCLUSION

In conclusion, The Marquardt Company R-4D rocket engine is an extremely reliable, versatile liquid bipropellant 100-lb. thrust rocket engine. An early model was used as a mid-course and orbit adjust engine on the highly successful NASA Lunar Orbiter program and, in one instance, was used for orbit adjust after being in lunar orbit for 335 days.

It has performed flawlessly in its intended modes of operation as reaction control engines on the Apollo Service Module and Lunar Module vehicles. In this capacity it performs the following functions:

- Provides CSM/S-IVB Separation
- Provides docking attitudes in the LM and LM ejection maneuvers
- Provides attitude control during mid-course corrections
- Provides thrust for rotisserie temperature conditioning roll for passive thermal control
- Maintains attitude during translunar coast and navigational sightings.
- Small mid-course corrections
- Orients spacecraft for SPS burn during lunar orbit insertion
- Orients and maintains spacecraft during lunar orbit.
- Orients and maintains attitude during any SPS burn and provides ullage for SPS tank
- Provides attitude control and propulsion for CSM/LM undocking and separation maneuvers
- Provides attitude control for LM decent and lunar landing
- Provides attitude control for LM ascent
- Provides attitude control and thrust during LM orbit adjustments for CSM rendezvous.
- Provides attitude control and thrust during CSM-LM rendezvous
- Provides for LM jettison during CSM-LM separation maneuver.
- Orients and maintains attitude control during SPS burn and provides ullage for SPS tanks during trans-earth injection

- Provides attitude control and ΔV translation and maintains roll for passive thermal control during the return to earth.
- Provides attitude control and thermal for CM and SM separation.

A total of 758 R-4D engines have been fabricated, acceptance tested and delivered to North American Rockwell Corporation for use on the Apollo Service Module and to Grumman Aerospace Corporation for use on the Apollo Lunar Module. In addition, several R-4D's have been delivered to NASA/MSC for use in ground testing. Eleven of the earlier models were delivered to The Boeing Company for use on Lunar Orbiter.

The engine is completely qualified and operates on helium saturated propellants: Aerozine-50 (MIL-P-27402) and MMH (MIL-P-27404) fuel and both "brown" (MIL-P-2653A) and "green" (MSC-PPD-2A) oxidizer.

The engine has also been pre-qualified (Design Verification Tests) with a Columbium chamber replacing the molybdenum chamber.

The engine historical test firing summary for both molybdenum and columbium chambers is presented below:

	ENGINE MODEL	RATED THRUST (lbs)	NO. OF ENGINES TESTED	NO. OF IGNITIONS	TOTAL ACCUMULATED FIRING TIME
MOLYBDENUM					
Total Engines Tested	R-4D	100	612	774,693	1.99 days
Individual Engine and Combustor Performance					
1. Maximum no. of engine ignitions by a single eng.	R-4D (T-12285)	100		103,548	
2. Maximum accumulated burn time by a single engine	R-4D	100			5.41 hours
3. Maximum accumulated burn time using a single comb.	R-4D	100			8.85 hours
4. Maximum continuous burn time by a single engine	R-4D (T-12321)	100			2.0 hours
COLUMBIUM					
Total Engines	R-4D	100	32	29,744	7.08 hours

	ENGINE MODEL	RATED THRUST (lbs)	NO. OF ENGINES TESTED	NO. OF IGNITIONS	TOTAL ACCUMULATED FIRING TIME
Individual Engine and Combustor Performance					
1. Maximum No. of ignition by a single engine	R-4D (228686-501, S/N 004)	100		7,051	2.14 hours
2. Maximum accumulated firing time by a single engine	R-4D (T-11606-001)	100		50	2.32 hours
3. Maximum accumulated firing time using a single combustor	R-4D (T-11606)	100		26	1.95 hours
4. Maximum continuous firing time by a single engine	R-4D	100			1.00 hours

The Model R-4D space firing summary as of July 24, 1969 is as follows:

MODEL R-4D SPACE FIRING SUMMARY

JULY 24, 1969

LUNAR ORBITER PROGRAM (FIVE FLIGHTS)

SPACECRAFT IDENTIFICATION	TOTAL ENGINE STARTS	TOTAL BURN TIME	SPACECRAFT TIME IN SPACE	ENGINE TIME IN SPACE (ENGINE DAYS)	NO. OF ENGINES
L. O. I	5	12.2 MIN.	80 DAYS	80 DAYS	1
L. O. II	7	12.5 MIN.	338 DAYS	338 DAYS	1
L. O. III	7	12.5 MIN.	243 DAYS	243 DAYS	1
L. O. IV	4	11.9 MIN.	176 DAYS	176 DAYS	1
L. O. V	6	12.5 MIN.	180 DAYS	180 DAYS	1
TOTAL TIME HISTORY FOR LUNAR ORBITERS	29	1.03 HOURS	2.79 YEARS	2.79 YEARS	5

APOLLO SPACECRAFT PROGRAM (TEN FLIGHTS)

MISSION IDENTIFICATION	SPACECRAFT IDENTIFICATION	TOTAL ENGINE STARTS	TOTAL BURN TIME	SPACECRAFT TIME IN SPACE	ENGINE TIME IN SPACE (ENGINE DAYS)	NO. OF ENGINES
APOLLO (AS201)	SC 009	1,818	4.1 MIN.	0.5 HRS.	0.3 DAYS	16
APOLLO (AS202)	SC 011	7,040	8.0 MIN.	1.5 HRS.	1.0 DAYS	16
APOLLO 4	SC 017	15,749	9.3 MIN.	8.5 HRS.	5.7 DAYS	16
APOLLO 5	LM 1	8,540	36.5 MIN.	8.0 HRS.	5.3 DAYS	16
APOLLO 6	SC 020	19,472	17.5 MIN.	9.6 HRS.	6.4 DAYS	16
APOLLO 7	SC 101	61,000	41.5 MIN.	10.8 DAYS	172.8 DAYS	16
APOLLO 8	SC 103	46,240	27.3 MIN.	6.0 DAYS	96.0 DAYS	16
APOLLO 9	SC 104	41,100	25.8 MIN.	10.0 DAYS	160.0 DAYS	16
	LM 3	25,230	17.4 MIN.	4.2 DAYS*	67.2 DAYS	16
APOLLO 10	SC 106	44,700	28.1 MIN.	8.0 DAYS	128.0 DAYS	16
	LM 4	34,650	23.9 MIN.	4.5 DAYS*	72.0 DAYS	16
APOLLO 11	SC 107	50,900	27.8 MIN.	8.0 DAYS	128.0 DAYS	16
	LM 5	16,950	12.1 MIN.	5.5 DAYS	88.0 DAYS	16
TOTAL TIME HISTORY FOR APOLLO R.C.S. ENGINES		373,389	4.65 HOURS	58.2 DAYS	2.55 YEARS	208

TOTAL SPACE FIRING TIME HISTORY FOR R-4D ENGINES USED ON APOLLO AND LUNAR ORBITER SPACECRAFT

ENGINE STARTS	TOTAL BURN TIME	SPACECRAFT TIME IN SPACE	ENGINE TIME IN SPACE	NO. OF ENGINES
373,418	5.68 HOURS	2.94 YEARS	5.34 YEARS	213

* TIME PERIOD FROM EARTH LAUNCH TO FINAL JETTISON OF LM ASCENT STAGE FROM COMMAND SERVICE MODULE.