



BELL AEROSYSTEMS COMPANY

EXPERIMENTAL AUXILIARY ROCKET ENGINE

CONTRACT NAS 7-305

FINAL REPORT

PART I

REPORT NO. 8374-933004

1-5-66



FOREWORD

Part I of this report includes the following:

Section

- I Introduction**
- II Program Progress Summary**
- III Program Plan**
- IV Summary of Technical Progress - Task I and II**
- V Technical Discussion - Task I**

Part II of this report includes the following:

Section

- VI Technical Discussion - Task II**
- VII Trips and Visits**
- VIII Conclusions and Recommendations**

Appendix I

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Credits

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I. INTRODUCTION

This final report summarizes the work accomplished by Bell Aerosystems Company under NASA Contract NAS 7-305. The schedule period for this program was from contract go-ahead on 19 June 1964 to 15 January 1966.

During this program, the main objective was to determine and demonstrate a thrust chamber design concept that was insensitive to duty cycle and one that could satisfy a wide spectrum of applications ranging from reaction control system engines to space-craft maneuvering engines. The effort was directed toward evolving and demonstrating the design concept at the 100-pound thrust level with storeable propellants. The concept should be suitable, however, for use at thrust levels from 5 to 1000 pounds and have growth potential to high specific impulse propellant combinations.

The adiabatic or insulated engine was selected as the concept that would best meet the requirements of this program. The feasibility of this concept was demonstrated on two prototype engines revealing the capability of this design to operate satisfactorily under continuous operation and pulse mode operation.

II. PROGRAM PROGRESS SUMMARY

As has been previously stated it was the object of the subject program to define and demonstrate a thrust chamber design concept which satisfies a wide spectrum of applications in reaction control and spacecraft maneuvering systems which will be suitable for installation either internal or external to the vehicle. The concept must be insensitive to duty cycle.

The concept selected was the adiabatic insulated refractory metal wall engine utilizing a bipropellant valve.

The prototype engines fabricated were of the 100 pound thrust size.

The following is a tabular summary of the very ambitious performance objectives or desired engine characteristics together with the measure of actual attainment during preprototype and prototype demonstration testing.

PERFORMANCE OBJECTIVES ACCOMPLISHMENT SUMMARY

<u>General Engine Characteristics</u>	<u>Prototype S/N 1</u>	<u>Prototype S/N 2</u>	<u>Development</u>
Thrust Level - 100 lb @ vacuum	100 lb	100 lb	
Propellant Selection	N ₂ O ₄ MMH	N ₂ O ₄ MMH	N ₂ O ₄ MMH
Installation - Insulated	X	X	X
Radiation			X
Operating Life - 3600 sec continuous	—	29 minutes	4 minutes (insulated) 17 minutes (total)
- 3600 sec pulsing	10.5 minutes	—	
- 50,000 valve cycles	4906	6	
Minimum Impulse Bit - 0.5 lb-sec	0.39 lb-sec	—	—
Min. Performance Steady State 290 sec	291.5 (measured) C _F = 1.78	291 (calculated) C _F = 1.78	293 (calculated) C _F = 1.75

PERFORMANCE OBJECTIVES ACCOMPLISHMENT SUMMARY (CONT)

<u>General Engine Characteristics</u>	<u>Prototype S/N 1</u>	<u>Prototype S/N 2</u>	<u>Development</u>
External Wall Temp. - 400°F max	200°F	386°F (1)	
Envelope Limitations 8 inch Diameter	O.K.	O.K.	
Suppress Ignition Spike			TF-5 Injector Shows Suppression Capability
Feed Pressure Max 300 psi (max)	< 200	< 200	

(1) Based on one fiberglass temperature only. Other temperature data compromised due to exhaust gas blowback in test cell.

III. PROGRAM PLAN

The program consisted of three major tasks as follows: Task I - Analysis and Design; Task II - Fabrication and Testing; and Task III - Documentation. The program was originally scheduled for a 12-month period from 19 June 1964 to 19 June 1965. Due to program modifications and problems encountered, the schedule was increased to a 19 month period with the completion date extended to 15 January 1966.

The original 12-month program plan is shown in Figure III-1; Figure III-2 shows the actual 19-month program plan.

The following paragraphs give a brief description of the effort accomplished during Tasks I, II and III.

A. TASK I - ANALYSIS AND DESIGN

During this task, design and analysis effort, supported by necessary laboratory tests, were conducted to establish an engine configuration which best satisfies the general requirements.

Revision "A" to the program plan, implemented in November of 1964 provided for a very limited investigation of the ignition "spike" problem and evaluation of an advanced state-of-the-art oxidation resistant coating. These program modifications resulted in a program schedule increase of 6 working weeks, with no additional definitized contract costs.

A Revision "B" program implemented in February 1965 extended the Task I effort from 7-1/2 months to 10 months. This plan discontinued the ignition "spike" evaluation previously implemented, allowed for an additional injector iteration, and provided for additional refractory metal/insulation compatibility evaluation.

The Task I effort consisted of:

- (1) Propellant selection for demonstration tests.
- (2) Establishment of chamber and injector design concepts and the analysis of thermal, mechanical, fluid dynamics and performance characteristics.

Go Ahead
June 19

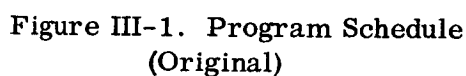
MONTHS AFTER GO-AHEAD

MILESTONES

NASA WESTERN OPERATIONS OFFICE REQUEST FOR PROPOSAL No. 64-14 EXPERIMENTAL AUXILIARY ROCKET ENGINE STATEMENT OF WORK		June	July	2
TASK I. ANALYSIS AND DESIGN A program of design and analysis, supported by laboratory tests where necessary, shall be conducted to establish the engine configuration that will best satisfy the general requirements. Summary of Work (a) The contractor shall conduct a comprehensive thermal, mechanical, fluid dynamic and performance analysis of candidate design concepts to establish the most attractive configuration(s). (b) The most severe duty cycle(s) for the selected configurations shall be determined analytically and each design examined under these conditions for conformance to requirements. (c) Where necessary, laboratory tests shall be conducted to provide meaningful inputs to the analytical programs. (I.e., injectors may be tested to establish thermal load to the engine, physical or thermal properties of materials may be examined, etc.) (d) The candidate configurations shall be examined for adaption over the 5 to 1000-pound thrust range, the growth potential to higher combustion temperature propellants shall be determined, and the throttling potential of the design shall be explored. (e) The flight weight engine configuration proposed for demonstration in Task II shall be submitted by the contractor and submitted to NASA for approval. (f) A demonstration program plan, including quantity of hardware, number and types of tests and facilities to be employed shall be developed by the contractor and submitted to NASA for approval prior to initiation of Task II.				
TASK II. FABRICATION AND TESTING Conduct a program of fabrication and testing to demonstrate the capability of the selected design concept under environmental conditions typical of those for spacecraft and re-entry vehicles, including the environmental conditions imposed by a typical Apollo launch vehicle. Emphasis shall be placed on demonstrating the engine characteristics under the most severe duty cycle(s) established under the Task I Analytical Program; however, other duty cycles shall also be demonstrated for correlation with the analytical predictions. (a) A minimum of three engines shall demonstrate conformance to the general requirements when operated under critical duty cycle conditions at simulated spacecraft and/or re-entry vehicle environmental conditions. (b) The instrumentation shall be adequate to establish the engine operating characteristics and to define the propellant feed system and vehicle interface requirements. (c) The data obtained from this program shall be used to verify the adequacy of the Analytical Program and if necessary shall be used to refine it.				
TASK III. DOCUMENTATION (a) The contractor shall provide documentation of all test condition and test results. Include in the final report the results of all key analyses and critical design details. (b) The contractor shall prepare and submit to NASA program plans, including schedules and cost estimates, for developing this design through qualification for both manned and unmanned applications. (c) A copy of each Analytical Program developed in conjunction with this project, including instructions and nomenclature, shall be delivered to NASA at the conclusion of Task II.				
SUMMARY OF PROPOSED PROGRAM (1) Review Program Requirements and Current Technology, and Make Propellant Selection. (2) Establish Candidate Injector and Thrust Chamber Design Concepts. (3) Perform Thermal, Mechanical, Fluid Dynamic, and Performance Analysis on the Candidate Concepts, Utilizing Data from 1c as Available - Principal Design Considerations - a. Injector Performance and Heat Rejection b. Thrust Chamber Cooling Concept for Exposed Installation (Radiation Cooling) c. Thrust Chamber Cooling Concepts for Buried Installation i. Passive Insulation on Refractory Metal Thrust Chamber ii. Active Coolant and/or Insulation iii. Operation During Reentry Conditions d. Other Design Considerations: Materials, Coatings, Thermal Shock to Injector Start Response, Minimum Impulse, System Dynamics, Propellant Utilization Off Design Limit Operation, Malfunction, Weight, Vibration, Particle Radiation Effect, Vacuum Operation, Fabrication Techniques e. Valve Design i. Biprop and Single Actuated Valves (4) Establish the Equations of System Dynamic Behavior in a Form Suitable for Analog Simulation and Determine Values of Parameters of Candidate Designs (5) Set up an Electronic Analog Simulation including the Necessary Function Generators, Transport Lags and Relay Circuits (6) Determine Dynamic Behavior of System Under Various Duty Cycles (7) Establish the Equations of System Heat Transfer as a Function of Environment and System Dynamics, and Determine the Values of Parameters for Candidate Design Prepare Item Transfer Equations for Solution on IBM 7090 (8) Determine Thermal Conditions of System Under Various Duty Cycles (9) Analyze Candidate Systems for Conformance with Requirements (10) Build Up Test Cell for Injector Tests (Cell B-2) (11) Active Coolant Tests (Model 8999 Columbian T/C) (12) Insulation Tests (Model 8999 Columbian T/C) (13) Design and Fabricate H ₂ O Cooled T/C (14) Design & Fab. Hot Wall Colum. Test Chamber (15) Colored Water Tests (16) Lev. Perf. & Heat Rev. Tests (a) No. 1 8 Element (Ball) (b) No. 2 8 Element (Film Cooled) (c) No. 3 8 Element (Unbal.) (d) No. 4 8 Element (Unbal. - Modified) (e) No. 5 8 Element (Unbal. - Modified) (f) No. 6 Unconventional (Special Doublet) (17) Coating Evaluation (Establish Operational Limits) (a) Storable Propellants Containing Oxygen (b) F ₂ and HF Oxidizers (See Note A) (18) Hot Wall Chamber Perf. Tests (Most Promising Injectors) (Two) (19) Hot Wall Chamber Pulse Tests (Most Promising Injector) (One) (a) Test Cell Build Up (X-E-East) (b) Fabricate Stainless Steel Chamber (c) Conduct Altitude Pulse Ignition Tests (20) Feasibility of all Welded Engine Assy. (21) Application of Concepts to Thrust Range 5-1000 lb (22) Application of High Energy Propellants (Note B) (23) Throttling Potential of Selected Components (24) Select and Perform Prototype Design (25) Design Review with NASA (26) Demonstration Program Plan for Task II (27) Redesign as Result of Design Review and NASA Review (28) Fabricate 4 Prototype Units (29) Prototype Engine Tests (All at 100,000 R min) a. Performance Tests b. Critical Duty Cycle Test c. Endurance and Operating Life d. Demonstration of Minimum Pulse Capability (30) Test Result Review and Correction with Analysis (31) Submit Monthly Progress Reports (32) Program Plan for Design Qualification (33) Submission of Analytical Programs (34) Final Report				

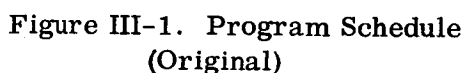
Establish Candidate Design Concepts
 Requirements
 Thrust
 Thermal, Mechanical
 VACATION
 Establish Heat Transfer Parameters and Properties
 Insul. Tests
 Design
 Modification of Engine
 Review Concepts
 Investigation of Application of High Energy Propellants
 Investigate Throttling
 Prepare and Submit
 VACATION
 Submit 1st Monthly Report
 Submit 2nd Monthly Report

III-2-1



Report No. 8374-933004

III-3-1



		Go Ahead June 15	
		MONTHS AFTER GO-AHEAD	
		June	July
		MILESTONES	
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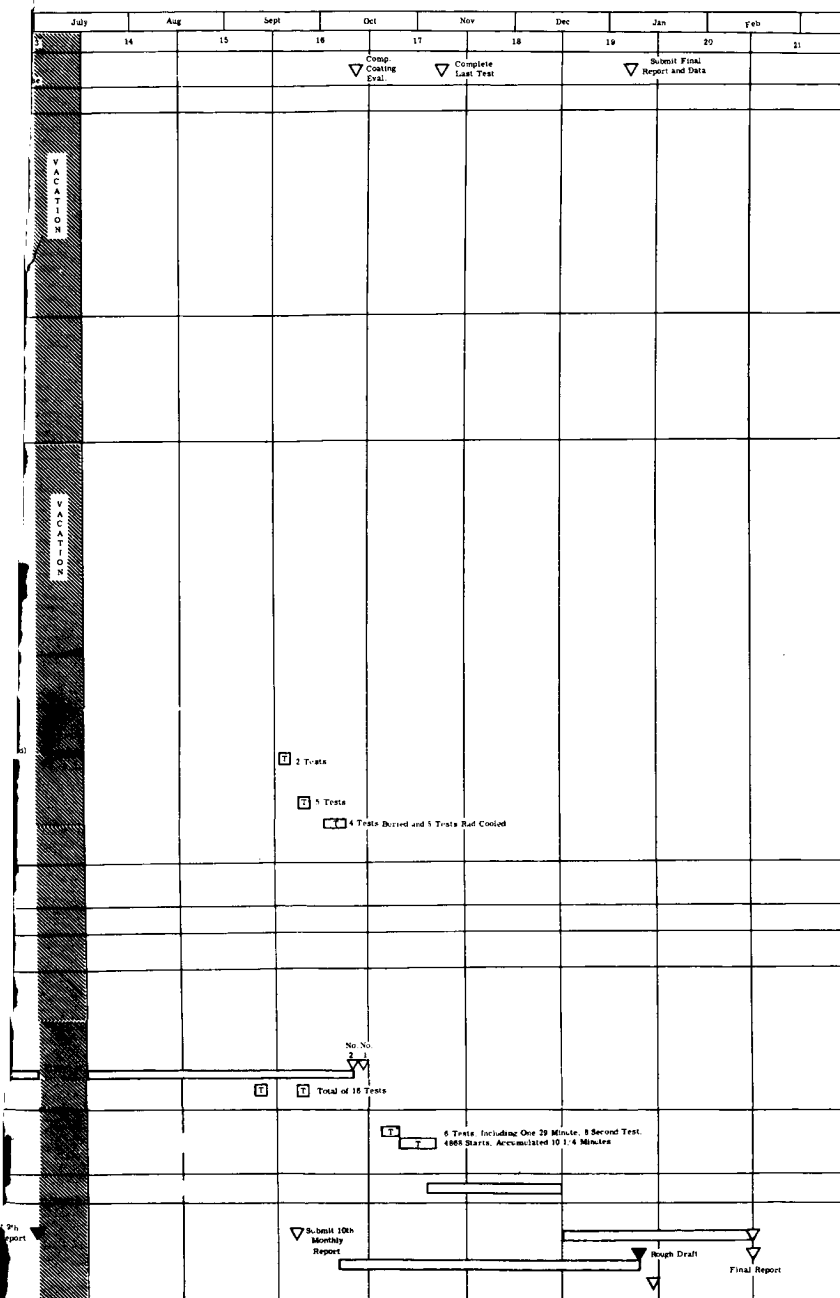


Figure III-2. Program Schedule
(As Actually Implemented)

IV. SUMMARY OF TECHNICAL PROGRESS

TASK I

A. PROPELLANT SELECTION

The performance values and general characteristics of the N_2O_4 and MON oxidizers and the hydrazine derivative fuels were evaluated. A numerical rating system was established to aid in selection of the propellant combination used for the demonstration testing. The propellants selected were N_2O_4 for the oxidizer and MMH for the fuel.

B. CHAMBER DESIGN CONCEPTS AND ANALYSIS

Detailed analyses were made to determine the best thrust chamber design concepts suitable for long duration (one hour) continuous operation and pulse mode operation for a buried installation.

Five basic thrust chamber design concepts were analyzed and are defined in the following paragraphs.

1. Ablative

Examination of the design concept with an 8-inch envelope limitation revealed that, with driving gas temperatures of $3500^\circ F$, a one hour continuous operation is nearly acceptable but pulse mode operation ~~cannot~~ be accommodated. Continuous operation for one hour results in an outside wall temperature of approximately $480^\circ F$ and pulse mode operation results in outside wall temperatures of approximately $700^\circ F$.

2. Thermally Thick Regeneratively Cooled (Liquid)

Analyses conducted on a refractory metal chamber radiating to a regeneratively cooled jacket shows the concept is acceptable for one hour continuous operation but is unacceptable for pulse mode operation. The coolant temperature ~~rise~~ becomes excessive during pulse operation. The heat that is rejected from a regeneratively cooled chamber must

be absorbed by the prppellant, therefore the heat capacity of the coolant must match the total heat rejected.

To reduce the heat rejected to the coolant to an acceptable value requires that the chamber L^* be reduced to approximately 3. The effect on performance with such a short L^* is undefined.

3. Radiation - Reflective Shield

Analysis shows a radiation cooled chamber enclosed in a reflector and depending on cooling through the annulus area and nozzle extension results in an excessive envelope size. With a 3500°F chamber wall temperature capability the envelope diameter limitation is only exceeded by less than $1/4$ inch. However, with a 2900°F chamber wall capability the envelope diameter limitation of 8 inches is exceeded by $5\ 1/2$ inches.

4. Radiation - Reflector Shield and High Axial Conductor

Adding an axial conductor $1/2$ inch thick along the chamber wall and using a thermal conductivity similar to that of silver improves the radiation reflection shield concept. With this type installation, the 8-inch envelope diameter limitation is exceeded by $2\ 1/2$ inches, assuming a 2900°F chamber wall capability.

5. Adiabatic

This concept is the most attractive due to simplicity of the design and ability to operate independent of duty cycle since the steady state, continuous operation is the most severe condition. Analyses have shown that within an 8-inch envelope the outside wall temperatures can be maintained below 400°F by use of a composite insulation system. The requirement for this concept is an injector capable of lower driving gas temperature or a chamber material, coating system capable of temperatures above 3500°F .

6. Fuel Rich Gas Regeneratively Cooled Concept

This cooling concept was considered somewhat advanced for this present program application but was analyzed to determine its feasibility since it has good throttling

potential and growth potential to higher energy propellants. The preliminary analysis reveals the envelope and outside wall temperature limitations can be met. The problems of developing this concept into a "working" piece of hardware, however, made this design incompatible with the schedule and budget limitations.

C. DYNAMIC BEHAVIOR ANALYSIS

Thermal equations and a mathematical model were established to analyze the effect of system dynamic behavior under various duty cycles. Attention was given to the model to determine its ability to produce a reasonable physical description of the pressure response which would be similar to test results. The model appears to be a good candidate for modification to incorporate effects of instability and adverse flow and pressure disturbances. Improvements to this model is planned under Bell sponsorship.

D. INJECTOR DESIGN CONCEPTS AND HARDWARE

1. Six injector configurations were established and evaluated during this program with the main objective being to develop an injector capable of generating driving gas temperature below 3000°F and delivering 290 seconds specific impulse (area ratio of 40:1). The injector identification is as follows:

<u>Designation</u>	<u>Drawing Number</u>	<u>Type</u>
T-1	8374-473001	Balanced triplet
TF-2	8374-473002	Film-cooled balanced triplet
TU-3	8374-473003	Unbalanced triplet
TS-4	8374-473004	Split triplet
TF-5	8374-473005	Unbalanced split triplet with preignitor and film coolant
TU-6	8374-473005	Unbalanced triplet

2. The basic test chamber design configuration was established with a nominal chamber pressure of 80 psia and an L^* of 15 inches.

Two segmented water cooled chambers were fabricated for the purpose of evaluating the various injector configurations for characteristic velocity, total heat rejection, circumferential heat flux distribution, and average heat flux in the chamber and nozzle section.

One stainless steel chamber was fabricated for use with the above injectors. A second stainless steel chamber designed for a chamber pressure of 135 psia was fabricated with Bell funding and also used with these injectors. The purpose of these chambers was to provide preliminary estimates of hot wall chamber heat rejection characteristics, obtain injector performance, and investigate vacuum ignition characteristics, including pressure spikes.

Two columbium alloy chambers (Cb - 1% zirconium) were fabricated for the purpose of determining injector performance under hot wall conditions, establish resultant steady state external surface temperatures under free radiating conditions and insulated or buried conditions, and to obtain more complete information for calculating driving gas temperatures adjacent to the combustion chamber wall.

3. Injectors T-1, TF-2, TU-3 and TS-4 were fire tested in the water cooled chamber. Injector TU-3 revealed the lowest heat rejection but also the lowest combustion efficiency. Film cooled injector TF-2 developed a low overall heat rejection but the film cooling proved inadequate at the throat station. Performance for this injector was satisfactory. Injector TS-4 proved to be the highest performing injector with corresponding high heat rejection.

4. Injectors TF-2, TS-4 and TF-5 were fire tested in the stainless steel chambers. Injector TF-2 was evaluated at chamber pressure levels of 80 psia and at 135 psia. Preliminary information regarding injector performance and heat rejection characteristics in a hot wall chamber was obtained for injectors TF-2 and TF-5.



Injector TS-4 was fire tested at 100,000 feet altitude on the 100-pound pulse mode stand to initially investigate the characteristics of the injector under pulse mode operation and to also check out the test stand.

Injector TF-5, which incorporates a spike suppression preignitor, was tested at simulated altitudes above 150,000 feet to determine its ability to reduce the pressure spike. Limited testing revealed the pressure spike was reduced by approximately 50% during the testing with the preignitor in operations.

An injector fabricated with Bell funding incorporating a "spike" suppression device (preignitor) was also fire tested at altitudes above 150,000 feet in an attempt to reduce or eliminate the "spike". A "spike" did occur during testing of this injector but was probably caused by an insufficient lead between the preigniter flow and the primary injector flow. Instability also occurred during the testing with the preignitor. Two preignitor elements and one precombustor element were also tested at altitude conditions to investigate the start characteristics and stability.

5. Injectors TF-2, TU-3, TS-4, TF-5 and TU-6 were fire tested in the columbium alloy radiation cooled chamber. Steady state performance and temperature data were obtained. The results of this testing revealed the unbalanced injector TU-6 provided chamber temperatures below 2400°F when radiating to space with calculated performance values above 293 seconds (at area ratio of 40:1).

Injector TU-3 was fire tested for two minutes in the columbium chamber under "buried" conditions using a columbium jacket insulated with alumina bubbles.

Injector TU-6 was fire tested in the columbium chamber for 4 minutes under "buried" conditions with alumina bubbles applied directly on the chamber. Measured chamber wall temperatures were less than 2800°F. These two test series proved the feasibility of the adiabatic concept and preliminary compatibility of the insulation material.

E. MATERIALS AND COATING EVALUATION

A vendor survey was conducted to determine the advanced state of the art for refractory metal coatings. The hafnium-tantalum coating system on tantalum tungsten was investigated revealing this type coating system was not developed sufficiently for application at this time. To investigate this type coating, two hafnium-tantalum clad chambers and one slurry applied hafnium-tantalum coated chamber were fire tested using injectors TF-2 and TU-6. Accumulated run time of 49 seconds was obtained on the first hafnium-tantalum clad chamber; 82 seconds were accumulated on the second at which times severe erosion was encountered. A total of 424 seconds was accumulated on the slurry applied hafnium-tantalum coated chamber, at which time severe throat erosion occurred.

Based on the survey of available coatings and experience at BAC, the columbium alloy SCb-291 was selected as the prototype chamber material coated with the GT&E modified silicide coating 508-C.

F. INSULATION EVALUATION

A survey of high temperature insulation materials was made, resulting in the selection of alumina bubbles, carbon fibers and Dyna Quartz as possible candidates for insulating the thrust chamber.

Temperature capability tests were conducted on graphite fibers and alumina bubbles at approximately 3500°F for 1 hour and at various pressures.

Laboratory tests were performed to determine the compatibility of these three insulators with coated and uncoated columbium, tantalum, tantalum-tungsten, and hafnium-tantalum. These tests were performed at various pressures and at temperatures to +3200°F for one hour.

The alumina bubbles were found to be compatible with the coated columbium alloy.

The alumina bubbles and Dyna Quartz insulation materials were selected as the composite insulation system.

G. PROPELLANT VALVE DESIGNS

A total of five propellant valve designs were reviewed and rated. Based on the results of this review and tests conducted by NASA-Houston, the Moog bipropellant valve was selected for the demonstration engine.

A throttling valve design manufactured by Fox Valve Inc. was evaluated.

A tabulation was made of possible propellant valve designs for thrust ranges between 5 pounds and 1000 pounds.

The review of an all welded propellant valve-to-injector interface concept was made.

H. PROTOTYPE ENGINE DESIGN

As a result of the various studies conducted the prototype engine concept selected was the adiabatic design. This design incorporated a columbium chamber and nozzle extension coated with a silicide 508-C coating, a columbium injector of the TU-6 configuration (unbalanced triplet), alumina bubbles insulation contained in a columbium shell, Dyna Quartz insulation around the columbium shell, an outerwrap of fiberglass, and a Moog bipropellant valve.

Thermal and structural analyses were performed on this design.

I. THROTTLING INVESTIGATIONS

A series of test firings were conducted on a 100-pound thrust chamber under a Bell sponsored program to investigate the throttling capability using a Fox throttle valve. These tests were made at sea level and at 125,000 feet altitude conditions. A throttling range of 4 to 1 was satisfactorily demonstrated with stable chamber pressure conditions throughout.

Additional throttling capability was demonstrated on injectors TU-3 and TU-6 using a Fox throttle valve.

J. TEST CELL BUILDUP AND TEST STRUCTURAL MODIFICATION

The sea level test cell B-2 and altitude test cell 2E-S and associated thrust test stand were modified to accommodate testing of the various injector concepts and the demonstration engines.

TASK II

A. FABRICATION OF PROTOTYPE ENGINE ASSEMBLIES

The drawings for the two prototype engines were released for fabrication on June 1, 1965. During the fabrication of both injectors difficulty was encountered in obtaining proper impingement. Both injectors were reworked then fire tested in a stainless steel chamber prior to welding in the thrust chamber. Fabrication of the 2 prototype engines were completed by October 8, 1965.

B. DEMONSTRATION TESTING

Prototype engine S/N 2 was subjected to a total of 6 tests at altitude including a continuous firing of 29 minutes and 8 sec. Following this firing, heat soakback was allowed until all major components had reached a peak temperature, then a 2 second firing was made to demonstrate restart capability. The intended one hour test was terminated at 29 minutes and 8 seconds due to a decrease of chamber pressure, indicating throat erosion or chamber burnthrough. Postrun inspection of the chamber revealed a burnthrough in the chamber section at approximately the 1:00 o'clock position. Throat erosion was also evident.

During these tests on Engine S/N 2 an aspirator duct was used to aid in maintaining 100,000 feet altitude for the long duration tests. The aspirator duct failed to maintain altitude after approximately two minutes of the test. Separation occurred in the chamber nozzle causing damage to the nozzle extension and resultant exhaust gas "blow-back" into the altitude chamber. Since the main objective of the test was to evaluate the coating capability, insulation capability with the coating, and the basic adiabatic design concept, the test was continued until chamber pressure indicated erosion.

Prototype Engine S/N 1 was then evaluated in the altitude facility at 100,000 feet simulated altitude. A total of 4906 starts and 10-1/2 minutes were accumulated on this assembly.

This series included 10 and 20 second duration tests to evaluate mixture ratio and chamber pressure effect, various pulse mode tests from 10 millisecond to 100 millisecond pulse, and a typical Apollo Command Module Duty Cycle. Visual inspection of this assembly following this duty cycle revealed the assembly to be in excellent condition. Additional 10 second duration tests were conducted until a loss of chamber pressure was noted.

A visual inspection of the assembly revealed a burnthrough in the chamber section at approximately 12:30 o'clock. Severe chamber erosion was noted.

Although neither prototype engine assembly demonstrated one hour capability, it is evident the adiabatic concept is feasible.

Although the life and temperature capability of the coating system is not clearly defined it is felt the life of each chamber would have been extended with improved injector impingement characteristics.

C. METALLOGRAPHIC INSPECTION OF CHAMBERS

A metallographic inspection of both chambers revealed the silicide coating provided good oxidation protection to the columbium and showed good compatibility with the alumina bubbles.

D. THERMAL ANALYSIS CORRELATION WITH TEST RESULTS

Although it was difficult to accurately compare the thermal data obtained from engine S/N 2 with the analysis due to the exhaust gas "blow back" into the altitude chamber, the temperature data that was reliable showed fair correlation with the analysis. The thermal data obtained from engine S/N 1 during pulse mode operation indicated good agreement with the analysis except in the region of the propellant valve.

E. PERFORMANCE AND INSTRUMENTATION

The measured performance for prototype engine S/N 1 revealed at a 1.6 mixture ratio, a characteristic velocity of 5280 ft/sec and a specific impulse of 292 seconds. The specific impulse for engine S/N 2 was approximately 291 seconds.

SECTION V. TECHNICAL DISCUSSION - TASK I

A. PROPELLANT SELECTION

A review and analysis was made of the general characteristics and performance values of the N_2O_4 or MON oxidizers and the hydrazine derivative fuels or mixtures to aid in selecting the propellant combination for demonstration testing. The general characteristics of these propellants are shown in Table V-1. The performance data for the various combinations are shown in Table V-2. These performance data are at a mixture ratio that yields the highest vacuum specific impulse at an area ratio of 40 and at a chamber pressure of 80 psia. Performance plots of the majority of these propellant combinations are shown in Figures V-1 through V-4.

A numerical rating system was also established for the oxidizers N_2O_4 and MON and for the fuels 50% N_2H_4 -50% UDMH, UDMH, and MMH. The remaining fuels were eliminated for the following reasons:

- (1) Hydrazine, although capable of delivering slightly higher performance was eliminated because of its low decomposition temperature and high freezing point.
- (2) The various hydrazine-MMH blends were eliminated because of the added complexities associated with the blends and the lack of sufficient data.

The results of this rating evaluation are shown in Table V-3 and reveals that N_2O_4 is favored over MON and MMH is favored over 50% N_2H_4 -50% UDMH, or UDMH. A summary of the advantages of these two propellants is given below:

MMH versus 50-50

- (1) Lower freezing point; may aid in reducing spike effect.
- (2) High boiling point; advantageous from chamber cooling concept.
- (3) Lower vapor pressure
- (4) Good specific heat

- (5) Good cooling capacity; decomposition temperature much higher than N_2H_4 in 50-50 blend
- (6) Lower combustion temperature at the rich (barrier) mixture ratios
- (7) No concern about reported non-miscibility with time and storage

N_2O_4 versus MON

- (1) Higher boiling point
- (2) Lower vapor pressure
- (3) Low compressibility
- (4) Easier propellant to analyze and more predictable since the NO content of MON can decrease with use and handling

Since the minimum vacuum specific impulse goal is 290 seconds, a propellant combination yielding the highest performance is most desirable. The propellant combination of N_2O_4 /MMH results in one of the three highest performance values of the combinations considered. Although the specific impulse is approximately one second less than that with N_2O_4 /50-50, the other advantages of using MMH overshadow this lower performance. Also, the performance versus mixture ratio curve is flat between the mixture ratio band of 1.5 to 1.9 which allows variations in mixture ratio with little effect on performance (See Figure V-1).

As a result of the above evaluation the propellant combination of N_2O_4 /MMH was selected as most desirable for the demonstration testing.

It should be noted that should additional chamber cooling be required and lower performance is permissible, the IRFNA/MMH blend is very attractive. This also results in a lower oxidizer freezing point which matches well with the MMH freezing point and may reduce spike effects. The purpose of blending MMH with N_2H_4 for this propellant combination is to increase performance by matching the fuel and oxidizer freezing point.

CANDIDATE PROPELLANTS

OXIDIZERS

	N_2O_4	MON 0.9 N_2O_4 0.9 NO	MDFNA 0.56 HNO_3 0.44 N_2O_4	IRFNA 0.85 HNO_3 0.15 N_2O_4	
Triple Point °F	+11.8	-11.5	-35	-56	
Triple Point Pressure psia	2.7	3.5	0.70	0.28	
Normal Boiling Point °F	+70	+41	+86	+139	
Vapor Pressure @ 77°F psia	17.7	33	11	2.5	
Density g/cc @ 77°F	1.43	1.41	1.62	1.55	
Specific Heat BTU/lb-°F	0.37	.	0.43	0.41	
Heat of Vaporization BTU/lb	178	158	153	297	
Heat of Fusion BTU/lb	68				
Viscosity μ lb/ft sec @ 77°F	2.65×10^{-4}		2.2×10^{-3}	8.3×10^{-4}	
Compressibility	Low	High	Low	Low	
Storability	Good	Good	Good	Good	
Stratification	None	None	Possible	None	
Coolant Capacity	Low	Low	Fair	Good	Ex De >

BELL AEROSYSTEMS COMPANY

E V-1

MODEL 8374 AUXILIARY ENGINE

FUELS

H_4	50-N $_2$ H $_4$ 50 UDMH	UDMH	MMH	Hydra- zine Diborane	BAF-8 0.75 MMH 0.25 N $_2$ H $_4$	BAF-9 0.86 MMH 0.14 N $_2$ H $_4$
.1	+21	-72	-63		-30	-64
05	0.30	0.02	0.10			
35	+170	+146	+189	Formed Solid Combustion Products		
278	2.8	3.26	0.96			
.00	0.899	0.785	0.874		1.07	1.08
.74	0.69	0.65	0.70			
603	426	251	377			
170		72	97.3			
0.05×10^{-4}	5.5×10^{-4}	3.4×10^{-4}	5.2×10^{-4}		2.3×10^{-3}	3.1×10^{-3}
Low	Low	Low	Low		Low	Low
Good	Good	Good	Good		Good	Good
Explosive composition 100°F	Fair	Fair	Good			

TABLE V-2
PERFORMANCE DATA

Performance ⁽¹⁾ data at mixture ratio yielding highest vacuum
specific impulse at an area ratio of 40 $P_c = 80$ psia

Propellant	M.R. (O/F)	$I_{sp\text{ vac}}$ theor.	$I_{sp\text{ vac}}^{(2)}$ 90%	c^* (ft/sec)	$C_{F\text{ vac}}$	T_c °R
N_2O_4 /50-50	1.55	321.8	289.6	5720	1.812	5435
N_2O_4 /UDMH	2.0	317.5	285.8	5640	1.810	5480
N_2O_4 /MMH	1.70	320.7	288.6	5700	1.810	5460
MON/BAF-7 ⁽³⁾	1.50	323.0	290.7	5740	1.810	5430
MDFNA/BAF-8 ⁽³⁾	1.70	316.8	285.1	5600	1.820	5280
RFNA/BAF-9 ⁽³⁾	1.90	312.5	281.2	5500	1.825	5200
RFNA/TMT	2.10	306.0	275.4	5400	1.822	5330

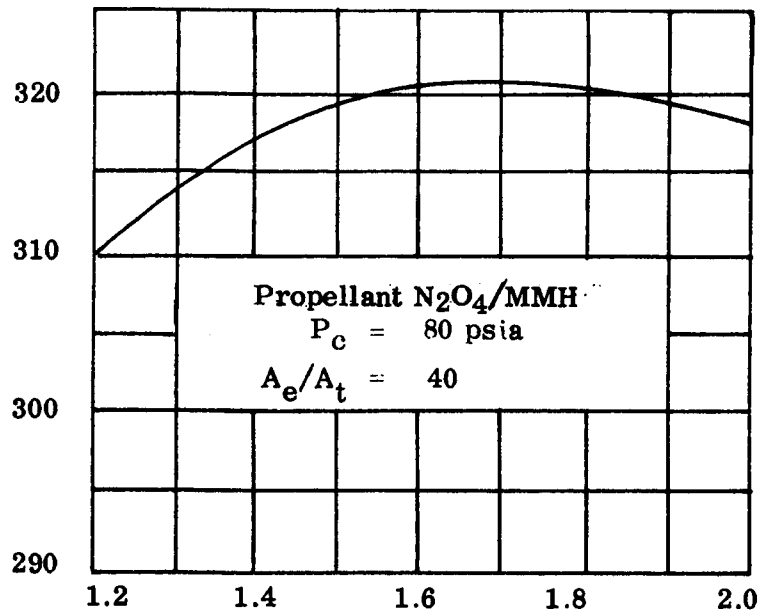
- (1) Performance based on equilibrium composition to throat station and frozen composition after throat
- (2) 90% specific impulse efficiency has been assumed as realistic basis for comparison of actual performance.
- (3) The BAF blends of fuel are various mixtures of MMH and N_2H_4 to match the freezing points of the fuel blend to the oxidizer freezing points and, by addition of the N_2H_4 , obtain an increase in performance.

TABLE V-3

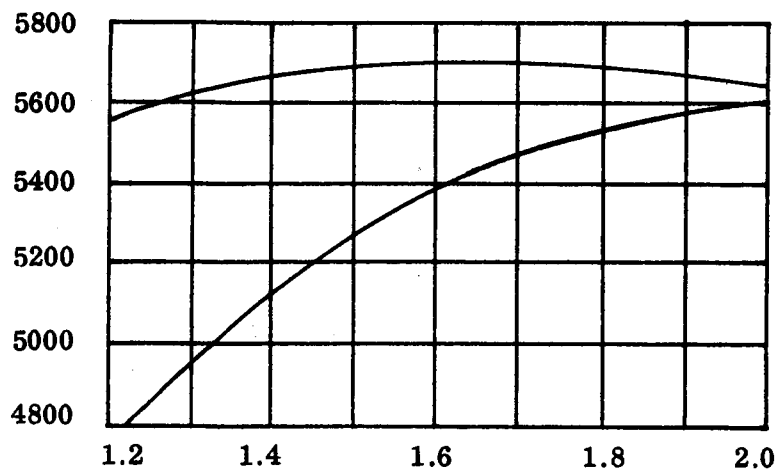
CANDIDATE PROPELLANTS, MODEL 8374 AUXILIARY ENGINE, NUMERICAL EVALUATION

Characteristic Evaluated	Value Assigned	Importance of Characteristic	Oxidizers				Fuels				
			N ₂ O ₄		MON 0.9N ₂ O ₄ -0.9NO		50 - N ₂ H ₄ 50 - UDMH		UDMH		MMH
			Rating +11.3	Rating -11.5	Rating +21	Rating -72	Rating +21	Rating -72	Rating +21	Rating -63	
Triple Point °F	10	Lower freezing point; may aid in reducing spike	5	10	5	10	5	10	5	10	
Triple Point Pressure psia	5	Lower freezing point; may aid in reducing spike	2.7	3.5	0.3	0.02	1	5	0.10	3	
Normal Boiling Point °F	10	High boiling point; advantageous from chamber cooling concept	+70	+41	+170	+146	8	5	+189	10	
Vapor Pressure @ 77° F psia	10	Lower vapor pressure simplifies handling procedures	17.7	33	2.8	3.26	7	6	0.96	10	
Density @ 77° F g/cc	5	Higher density gives more delivered impulse for same system volume	1.43	1.41	0.899	0.785	5	4	0.874	5	
Compressibility	10	Low compressibility, better the dynamic characteristics, i.e., stiffer hydraulic system	Low	High	Low	Low	10	10	Low	10	
Coolant Capacity	5	Higher coolant capacity, more suitable for variety of coolant concepts	Low	Low	Fair	Fair	3	3	Good	4	
Totals	55		(46)	38	39	43	(52)				

$I_{sp\ vac}$
lb_f - sec/lb_m



c^* ft/sec
 T_c °R



$C_{f\ vac}$

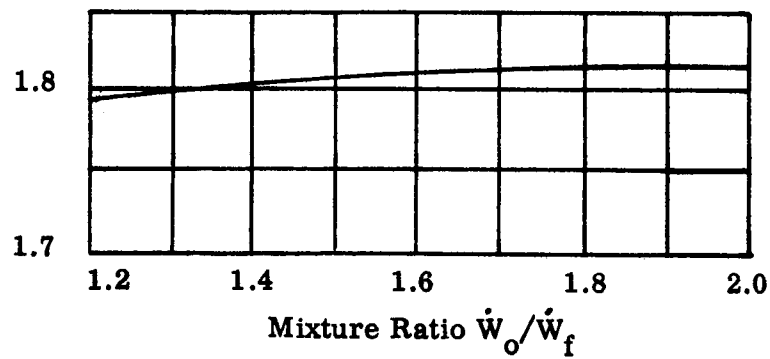


Figure V-1. Performance Curves

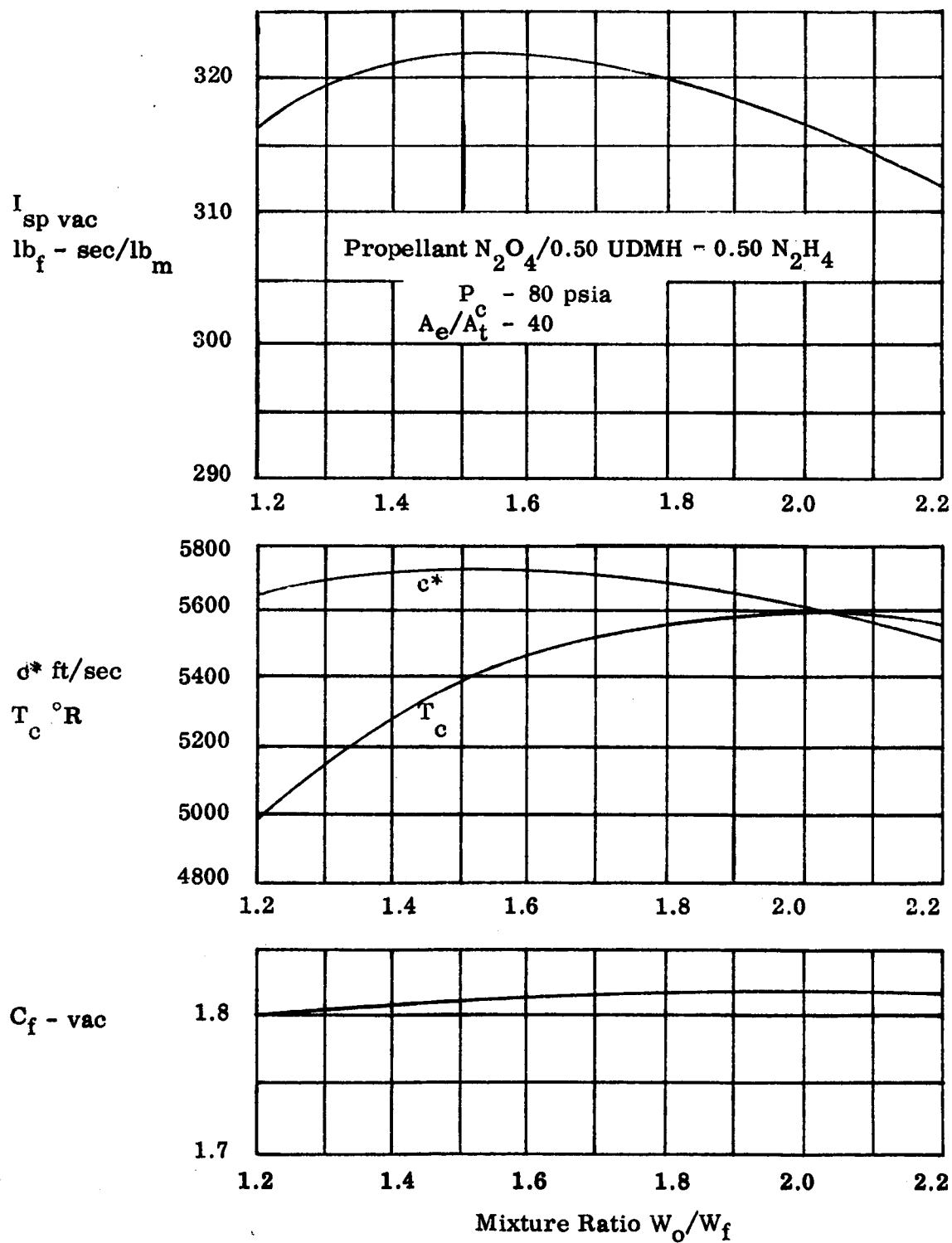


Figure V-2. Performance Curves

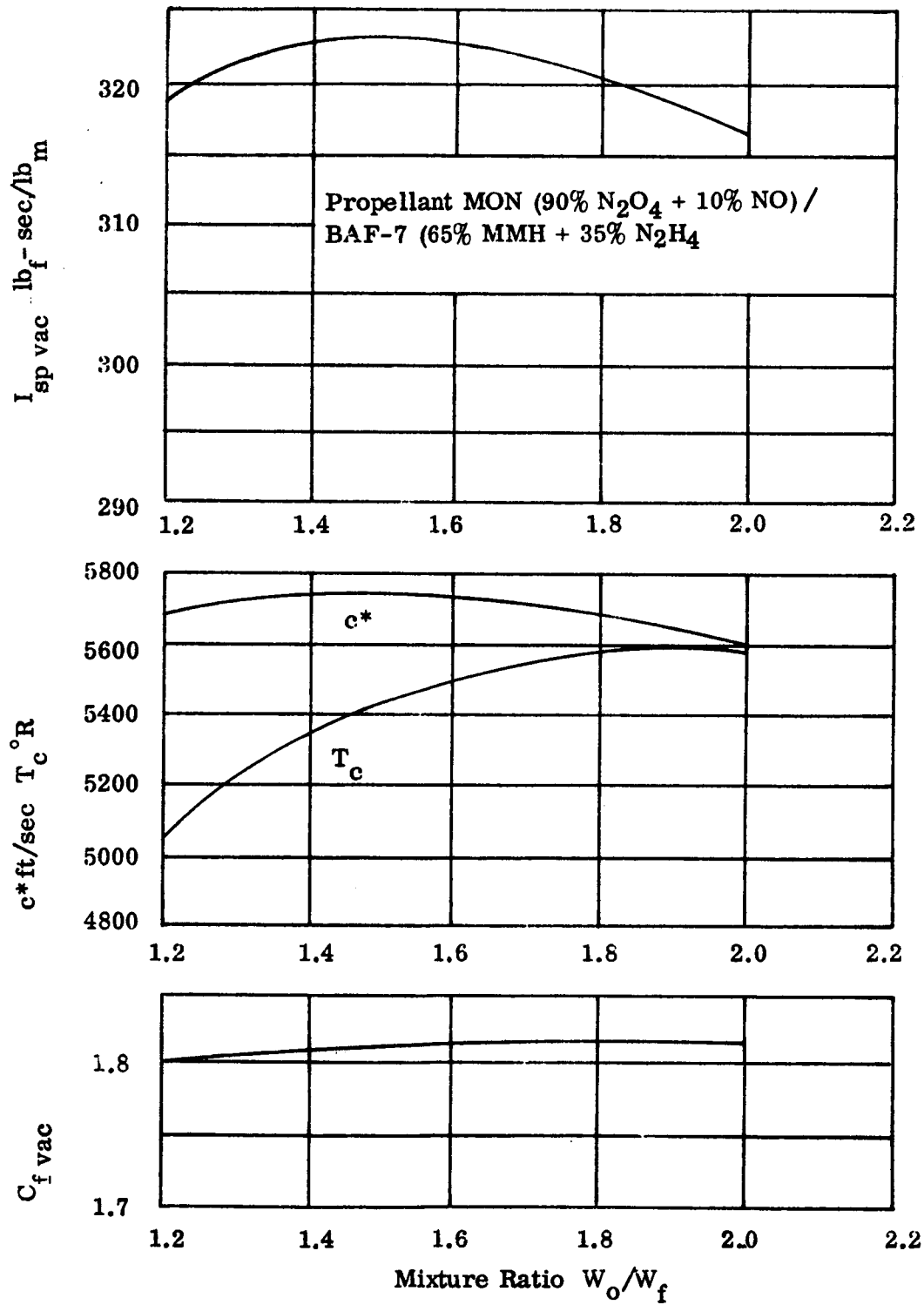


Figure V-3. Performance Curves

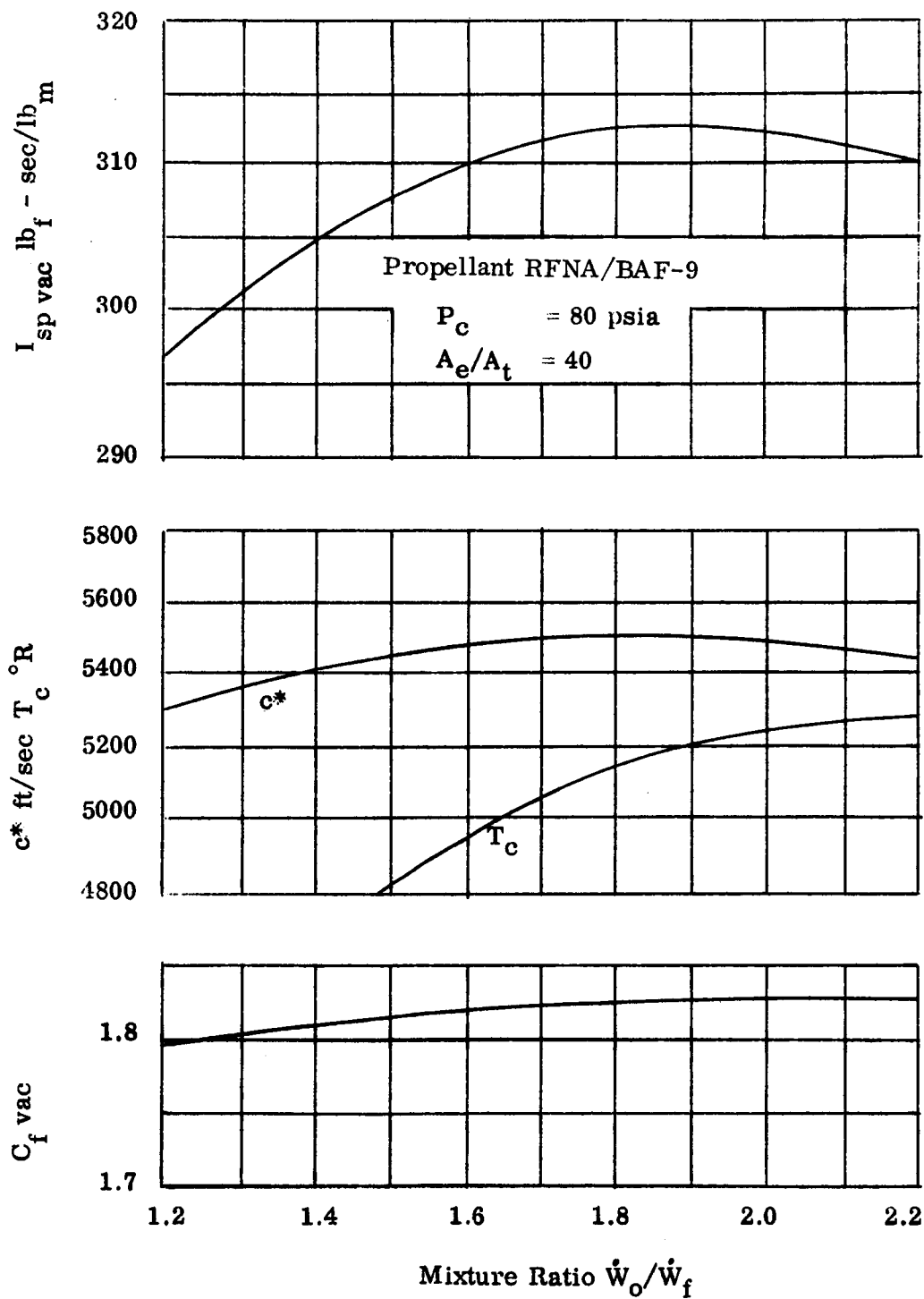


Figure V-4. Performance Curves



B. CHAMBER DESIGN CONCEPTS AND ANALYSIS

Initially a study of the candidate thrust chamber design concepts was made to determine the most attractive means of achieving low external wall temperatures of +400°F under buried installation. These concepts included the following:

- (1) Nearly adiabatic wall
- (2) Thermally thick wall plus convection cooling
 - (a) Insulated refractory chamber
 - (1) Propellant cooled
 - (2) Additional capacitor to absorb heat soakback
 - (b) Radiation cooled refractory chamber with a regenerative cooling coil
 - (c) Ablative matrix with a regenerative cooling coil
- (3) Radiation cooled with reflector

General analytical studies were conducted to include arrays of parameters which would encompass rockets of thrust levels between 5 and 1000 pounds. Results of these studies were such that a definition of significant functionalities of size, insulating capabilities, and gas temperatures could be made. Results of the work are expressed in figures which will be described in the following paragraphs:

1. Analytical Approach

The analysis initially included two basic concepts of rocket thrust chambers.

a. Insulated Thrust Chambers (nearly adiabatic)

This type of thrust chamber consists of thin wall refractory metal chamber, surrounded by a high resistance insulating shell, thus operating nearly adiabatic. A small amount of heat is conducted through the insulation and dissipated at the outer surface by radiation. The heat balance is specified by equation (1).

$$\frac{T_r - T_o}{\left[\frac{R_1 + t}{R_1} \right] \frac{1 + R_1}{h_g} \frac{\ln \left[\frac{R_1 + t}{R_1} \right]}{k}} = \epsilon \sigma T_o^4 \quad (1)$$

where:

- R_1 = insulation inner diameter, inches
- t = insulation thickness, inches
- T_r = gas temperature, °R
- T_o = external surface temperature, °R
- h_g = gas coefficient of heat transfer, BTU/in.²-sec/°R
- k = insulation conductivity BTU/in.-sec/°R
- σ = radiation constant
- ϵ = external emissivity

An IBM computer is used to produce solutions for arrays of chamber diameter, gas film coefficient, gas driving temperature, thermal conductivity of insulating material, and thickness of insulating material. The gas film coefficient is a function of the chamber pressure and physical size, both of which are related to an actual thrust chamber as specified by thrust level. The solutions to equation (1) are then examined in the following paragraphs.

b. Thermally Thick Wall Plus Convection Cooling

A thermally thick walled regeneratively fuel cooled thrust chamber was examined by considering varying degrees of thermal resistance between the gas side chamber wall and the coolant jacket. The thermal resistance reduces the amount of heat rejected to the coolant. The effect of thrust level, chamber pressure, and chamber radius have been examined to determine the suitability of regenerative cooling. The following relationship is used for defining the temperature rise of the coolant for a given thrust chamber.

$$\Delta T_f = \frac{A_s}{\dot{W}_f C_{p_f}} \left[\frac{T_r - 860}{1/h_g + t/k} \right] \quad (2)$$

where:

- A_s = surface area of jacket, in.²
- ΔT_f = temperature rise of fuel in cooling jacket, °R
- $\dot{W}_f$ = fuel flow rate, lb/sec
- C_{p_f} = specific heat of fuel, BTU/lb-°R

The thickness of the walls were sufficiently thin that only a one-dimensional case was considered.

2. Description of Results

Figures V-5 through V-12 represent the thermal aspects of families of thrust chambers. A 100-pound thrust chamber which has demonstrated a steady-state radiation cooled wall temperature of 2700°F was used as a reference point. This engine was shown to have had a driving temperature of the combustion gases of 3500°F and can be considered a rational design point.

a. Insulated Thrust Chambers

Figure V-5 indicates the variation in wall temperatures when the conductivity and material thickness is varied. Very small amounts of insulation will drop the outside wall temperature significantly, and the gas side wall temperature quickly approaches the adiabatic case. Within a prescribed outside envelope, say 4.0 inches, the outside wall temperature would be approximately 400°F, but the gas side wall temperature would be close to 3500°F. The unique line of 2700°F for both inner and outer walls represents the radiation cooled thrust chamber without insulation.

Figure V-6 indicates the strong dependency of the gas side wall temperature of the driving temperature. The thickness of insulation and the conductivity have very little affect on the gas side temperature while the outside wall temperature

will vary significantly with both thicknesses of insulation and conductivity of insulation. Reducing the insulation thicknesses raises the outside wall temperature because it would also increase the heating rate, however small.

Figure V-7 is indicative of the small dependency of chamber pressure and size as far as the operating temperatures are concerned. Examination of equation (1) shows that as the driving temperature potential and heat flux approach zero, the gas film coefficient may be allowed to vary over a wide range with little change in heat flux.

Figure V-8 leads one to the same conclusion as did Figure V-7, which shows the weak dependency on chamber pressure and the very strong dependency on gas driving temperature.

Figure V-9 examines the limits of radiation cooling when applied to a thrust chamber of low heating rates. Essentially, an emissivity of unity is the maximum limit where an emissivity of 0.7 is reasonably obtained in practice. The outside wall temperature is affected by the emissivity while the inside wall temperature is still controlled largely by the driving temperature.

b. Thermally Thick Regeneratively Cooled Thrust Chambers

Figure V-10 shows the temperature rise of the available coolant in a constant thrust size engine of the 100-pound thrust class. Various amounts of interspersed insulation are included so that relatively small amount of coolant temperature rise results. The extreme right hand margin would approach the adiabatic case where the gas side wall temperature increases approaching the driving temperature. The upper left corner approaches the maximum driving potential (3500°F) with an infinitely thin insulation. This would be an over-design point for this concept. The area of interest lies between the two superimposed lines of constant wall temperatures of 2600 and 3200°F . It may be shown that the knee of the curves is where the conductivity of the insulating material has a significant effect on the heating rate.

Figure V-11 is a plot similar to Figure V-10 except the physical size of the engine was varied. One can then observe that the larger the outer diameter, the greater total amount of heat will be rejected. Figure V-11 is designed around the 100-pound thrust engine where the typical wall radius is 1.0 inch.

Figure V-12 illustrates the passage size of a tubular type jacket to accommodate various thrusts for a representative coolant velocity. The tubes in the outside diameter show a weak functionality with thrust level. The range of interest on this plot lies between the two dashed lines of constant contraction ratio and would, with all probability, include those designs for engines of the 80 psia level. The constant coolant velocity of 15 ft/sec was assumed adequate for any cooling load.

It should be noted that the analysis described above is somewhat general and lacks detail. More detailed analyses were made with emphasis on pulse mode operations and steady state operation for applicable point designs within the above framework.

3. Critical Logic

The three basic modes of heat transfer; conduction, convection and radiation were further defined into five basic thrust chamber design concepts which are suitable for long duration or pulse mode firings for a buried installation. The five basic concepts are depicted schematically in Figure V-13. A short review of these concepts follows:

- (a) Ablative - This concept is sacrificial in nature, thus independent and therefore duty cycle dependent unless it chars to such an extent that it can operate steady state as a thick walled nearly adiabatic chamber that radiates.
- (b) Thermally Thick (Regenerative) - This concept is a regeneratively cooled chamber with a wall of high thermal resistance to reduce the heat rejected to the coolant. The wall may be constructed of the same material as an ablative or consist of a refractory wall with insulation.
- (c) Radiation (Shield) - This concept has a reflector shield of sufficient exit diameter so that reflected radiation cooling is passed aft around the gas restraining walls. An insulation is placed over the shield so as to present a low outside temperature to the vehicle enclosure.

- (d) Radiation (High Axial Conductor) - This concept provides peak wall temperature relief to the gas walls. Radiation cooling is provided by radiation aft through the nozzle exit. Again an insulator is placed over the conductor for installation requirements.
- (e) Adiabatic - This type of design concept is one with almost no heat rejected. The gas side wall is nearly of equivalent temperature to the combustion gas and a suitable layer or layers of insulation are applied to decrease the outside wall temperature to 400°F .

A network is presented in Figure V-14 which identifies the different types of concepts in greater detail, their potential and their problems. Depending on driving gas temperatures various design combinations appear suitable, others are eliminated for reasons indicated on the network.

For further identification of the design concepts, Figures V-15 through V-19 reveal pictorially five chamber concepts including basic temperature profiles across the chamber wall. Included is the basic radiation chamber, the radiation chamber with heat shield, the regenerative (semi) chamber with refractory wall, the regenerative (semi) chamber with an ablative wall, and the adiabatic wall chamber, respectively.

4. Analytical Studies

a. Ablative

The ablative matrix possesses thermal conductive resistance of such a nature that, coupled with out-gassing of resin and thermal capacity, a thickness of 3 inches on a 2-inch diameter chamber would operate steady state with a back side (outer) temperature of 480°F , above that of the specification limit of 400°F . Figure V-20 is a result of a transient analysis of such a phenolic matrix which radiates to space. As the duty cycle (time on to total time) is reduced, the outer wall temperature, for a constant total on time of 1 hour, unfortunately increases until a duty cycle of 0.065 is reached. As the duty cycle is further decreased, the outer wall temperature decreases.

For duty cycles greater than 0.065, the ablative wall is acting as a thermal capacitor; i.e., a steady-state condition has not been reached. However, for duty cycles less than 0.065, the ablative wall reaches a steady-state condition, i.e., the heat transferred from the gas to the inner wall is equal to the heat radiated from the outer surface. Thus, an inherent shortcoming in pulse mode is noted.

Figure V-21 shows the surplus heat as a function of duty cycle, which must be absorbed for a total firing time of 1 hour above that of a 400°F outer wall. Absorption of the excess heat could be accomplished by cooling with fuel, but heat soak-back and poor start transients would limit this even though the fuel available for a 1 hour firing has a capacity of 73,000 BTU.

A true, steady-state system under ideal conditions is shown on Figures V-20 and V-22. Here, in Figure V-22, after a time far in excess of 1 hour with no temperature gradient, is illustrated further the pulse mode requirements of a super-imposed liquid cooled system, or possibly, a capacitor type. The ablative system works as a radiation cooled thick wall cylinder and for continuous pulsing, a backface temperature of 750°F would be charred through since current matrix pyrolysis occurs at approximately 700°F. Thicker walls, as in Figure V-22 meet the temperature specification but with an outside diameter far in excess of 8 inches.

b. Thermally Thick (Regenerative)

(1) Steady-State

A critical review was made of a classic regenerative cooling design of a small thrust rocket. The following information summarizes the results and conclusions drawn.

Two materials were selected for a drilled configuration: tantalum tungsten and aluminum alloy. The first was selected with the intent of operating with thick walls so as to offer additional thermal resistance, thus reducing the rejected heat. It was possible to cool the tantalum unit with the fuel alone at 80 psia. The aluminum

design could not be operated even when cooling with both propellants. The steady-state characteristics are listed in Table V-4.

Fabrication techniques were strained beyond the state-of-the-art because of the small passage holes required to be grouped about the throat. One technique for maintaining a large number of small holes was suggested as the drilling of axially segmented sections or "wafers" which would be drilled and then electron-beam welded into a complete structure.

The fabrication technique would be formidable enough but the final stabilized temperature of the heavy structure with small trapped quantities of fuel would require a "dry jacket" with specific venting techniques. Pulse mode operation would be, response wise, poor.

TABLE V-4
REGENERATIVELY COOLED THRUST CHAMBER

Geometry

Nozzle Area Ratio	40:1	
Chamber Pressure, psia	80.0	
Throat Area, sq. in.	0.22	
Characteristic Length, inches	60.0	
Mixture Ratio	1.5	
Propellants	IRFNA + MMH	

Heat Rejection Characteristics

Materials of Construction	Tantalum	Aluminum
Fuel Flow Rate, lb/sec	0.0492	0.0492
Oxidizer Flow Rate, lb/sec	0.0731	0.0731
Heat Rejection Rate BTU/sec	4.83	13.15



TABLE V-4 (CONT)

Applicable Heat Capacity

BTU/sec

Fuel	5.84	5.84
Ox	3.12	3.12
Total	8.96	8.96

(2) Pulse Mode Operation

The pulse mode characteristics of a fuel cooled thermally thick unit was examined for pulse mode operation and the results are shown on Figure V-23. A consistent base line case of a 2.0 inch chamber with a driving temperature of 3500°F was coupled with an insulation thickness of 0.02 inches of an insulating material whose thermal conductivity was $1.0 \frac{\text{B in.}}{\text{ft}^2 \text{ hr}^\circ \text{F}}$. This work is an extension of the steady state work previously reported for this type configuration.

As the duty cycle is decreased, a decrease in wall temperature is noted as well as an increase in heating rate until that place where a radiation cooled thick wall cooling predominates. A maximum available fuel heat capacity of 73,000 BTU for a 1 hour firing is indicated by the vertical dashed line.

It is obvious that a very close margin of duty cycle (0.46) restricts the application of the thermally thick unit until a lower limit (0.00085) is reached. Between these limits a broad range of duty cycles are denied the propulsion system.

Also, it is obvious that the coolant temperature would be excessive if an upper limit of 300°F was honored to preclude chemical decomposition of the hydrazine based fuel coolant.

Assuming a maximum temperature increase of 200°F for the fuel the heat capacity of the fuel for a total of one hour of firing time is approximately 70,000 BTU. The total amount of heat rejected for the worst duty cycle of 0.001 is approximately 156,000 BTU for an emissivity of 0.7. Assuming a maximum temperature increase of 300°F

(or 400 to 460°F fuel temperature at maximum end of temperature band) the total heat capacity of the fuel is approximately 105,000 BTU. Vaporization of the fuel can be expected in this temperature range, resulting in possible performance changes that are not defined at this time.

Reduction of the surface area from 39 in.² to approximately 17.5 in.² would result in successfully matching the 70,000 BTU coolant capacity to a total rejected heat of 70,000 BTU. This results in a decrease of L* from 15 to approximately 3 and an undefined effect on performance. The details of this analysis are as follows:

The heat that is rejected from such a regeneratively cooled chamber must be absorbed by the propellants. The amount of heat rejected is directly a function of the value of the effective emissivity of the thrust chamber-jacket combination. If the emissivity is large, say 0.7, a significant reduction in the wall temperature from the driving temperature can be effected, say from 500°F to 1000°F depending upon the chamber pressure in question. Such an engine can be operated steady state but cannot in pulse mode because the heat rejection is too great. It is necessary to reduce the emissivity to one half that value or about 0.35 in order that the pulse mode operation can be accommodated. If the wall temperature is to be maintained at approximately 2700°F to guarantee satisfactory coating performance, the driving gas temperature must be decreased to approximately 3100°F. Such an operation is that much closer to an adiabatic condition.

The following relationships have been used:

$$h_g (T_g - T_{wg}) = \epsilon \sigma T_{wg}^4$$

where $h_g = 0.0003 \frac{\text{BTU}}{\text{in.}^2 \text{ sec } ^\circ\text{F}}$ at $P_c = 80$, duty cycle of unity

and h_g varies as $\left(\frac{\text{on time}}{\text{total time}} \right)^{0.8}$

This establishes the functionality of T_{wg} with duty cycle as shown in Figure V-24.

This functionally is completely appropriate if the duty cycle pulse bits are short, less than 1 sec "on-time". If the "on-time" is longer, the situation would be somewhat different and it would be necessary to examine it in more detail, but for this universal engine the case that is represented here is a valid mission to consider.

The heat that is to be picked up by the cooling system then is given as follows:

$$\text{total time } (A \in \sigma T_{wg_{\max}}^4) = Q_{\max}$$

We have found experimentally with a water jacket placed around the chamber that the average heating rate is about 0.7 the maximum rate associated with the maximum temperature so that we have adjusted the $Q_{\max}$ in that manner

$$\bar{Q} = 0.7 Q_{\max}$$

Now, knowing $T_{wg_{\max}}$ as a function of duty cycle, one can determine the heat that must be absorbed by the cooling system which is presented functionally in Figure V-25. This heat flux has been reduced by the amount that can be radiated from a 400°F outer surface. A tabulation of the data supporting this curve is shown in Table V-5.

The surface area of this chamber to an area ratio of 7 is 39 square inches.

The available coolant that has been determined is approximately 70,000 BTU.

This results from the fact that

$$\begin{aligned} Q_{\text{available}} &= \dot{W}_{\text{fuel}} \times C_p \times \text{time} \times \text{temp rise} \\ &= 0.1325 \frac{\text{lb}}{\text{sec}} \times 0.74 \frac{\text{BTU}}{\text{lb}^\circ\text{F}} \times 3600 \text{ sec} \times 200^\circ\text{F} \\ &= 70,000 \text{ BTU} \end{aligned}$$

TABLE V-5

HEAT REJECTED VERSUS DUTY CYCLE DATA

Duty Cycle	$T_{wg_{max}}^{\circ F}$	$\epsilon T_{wg_{max}}^4$ (BTU/ft ² hr)	$0.7 \epsilon T_{wg}^4$ - $\epsilon (860)^4$ (BTU/ft ² hr)	$39 \text{ in.}^2/144 (\text{in.}^2/\text{ft}^2)$ X 4
		<u>$\epsilon = 1.0$</u>	<u>$\epsilon = 1.0$</u>	<u>$\epsilon = 1.0$</u>
1.0	2700	173,000	120,160	32,543
0.3	2300	100,000	229,970	62,283
0.1	1930	56,500	386,100	104,239
0.03	1530	27,200	602,730	163,239
0.01	1160	11,800	732,000	198,250
0.003	830	4,790	804,253	217,820
0.001	640	2,520	824,000	223,164
		<u>$\epsilon = 0.7$</u>	<u>$\epsilon = 0.7$</u>	<u>$\epsilon = 0.7$</u>
1.0	2700	121,100	84,112	22,780
0.3	2300	70,000	160,979	43,598
0.1	1930	39,550	270,270	73,198
0.03	1530	19,040	421,911	114,268
0.01	1160	8,260	512,400	138,773
0.003	830	3,353	562,944	152,464
0.001	640	1,764	577,000	156,270

An imaginative solution would be a variable resistance unit wall where the gas side wall temperature would remain at a more constant, high level. A resulting independence of total rejected heat on duty cycle would make pulse mode operation feasible.

The mechanical challenge of such a variable resistance hot wall could possibly be met by servo-actuated concentric perforated sleeves which are axially shifted to allow innumerable combinations of registering holes so as to admit the proper amount of heat to the surrounding outer fuel cooled jacket.

c. Radiation (Shield)

External radiation coupled with internal radiation can be accomplished by surrounding the hot walls with a shield and providing an annular exit area. Increasing the exit annular area would lower the wall temperature until, in the limit, no inclosing is used and hence, the unit would operate totally as a pure radiation cooled unit.

Various geometric shapes were investigated: cylinders, frustrums of cones, and cylinders attached to frustrums of cones. For a given annular gap, it was found that the cylinder reflected the radiant heat in a more efficient manner than the other shapes and hence was used for the following work. Table V-6 is a summary of pertinent comparative temperatures for common parameters.

Figure V-26 presents results of this type of cooling with a shield of cylindrical geometry with insulation for a 400°F outer wall. Starting with a downward sloping line representing no exit annular opening, said opening is increased with resulting lower wall temperatures. The topmost curve is with no axial conductor. The highest temperature (with no gap) is the gas temperature. Addition of 0.5 inch of high axial conductor lowers the maximum wall temperature, again upstream of the throat.

The major radius would always occur at the exit radius to which would be added radial increases for:

Conductor

Annular gap for radiation

Insulation

The exit radius is the limiting dimension: The operating lip with cooling, less the outside wall temperature (400°F), divided by the radiant heat flux of a 400°F wall defines the resistance of the insulation in the shield. A conductivity of $0.5 \frac{\text{BTU in.}}{\text{ft}^2 \text{ hr } ^{\circ}\text{F}}$ for "Dyna Quartz" was used.

From Figure V-26 one may observe that the exit radius exceeds the envelope radius of 4.0 inches. This limitation could obviously be relieved by a reduction in combustion gas driving temperature.

TABLE V-6
EFFECT OF SHIELD CONFIGURATION ON THRUST CHAMBER
ASSEMBLY TEMPERATURES

Shield Geometry	Maximum Temperature ($^{\circ}\text{F}$)	
	Shield	Chamber
Cone (semi-angle = 10°)	2815	3500
Cone (semi-angle = 4.5°)	2565	3423
Cylinder	2427	3359
Cylinder and cone (semi-angle = 4.5°)	2474	3356

Exit Diameter of Shield = 9.0 inches

Emissivity Inside of Shield = 0.7

Emissivity Out Exit Plane = 0.2

d. High Axial Conductor (Radiation)

One type of passive cooling is that of enclosing a heat resistant wall with a superb conductor. Cooling through the exit plane is accomplished by radiation for which an emissivity of 0.2 is used.



Figure V-27 is a comparative example of adding various thickness of such a material to distribute or smooth out high points. The material was assumed to have a conductivity of $2000 \text{ BTU in./ft}^2 \text{ hr } ^\circ\text{F}$. For a thickness of 0 inches or an adiabatic condition, a wall temperature of 3500°F exists upstream of the throat. Cooling exists through the exit to drop the divergent nozzle wall temperature to 1840°F at the exit station.

Addition of a conductor with a thickness between 0.5 and 1.0 inches would lower the peak temperatures to those levels at which a state-of-the-art oxidation resistant coating would endure for extended periods of time.

The same cooling could be accomplished by a lower conductivity material if the thickness of this material is chosen such that the product of conductivity times the cross section area is held constant. Hence, a material of $1000 \frac{\text{BTU in.}}{\text{ft}^2 \text{ hr } ^\circ\text{F}}$ 2 inches thick is equivalent to 1 inch of $2000 \frac{\text{BTU-in.}}{\text{ft}^2 \text{ hr } ^\circ\text{F}}$ material (as on Figure V-27).

Investigation of a higher emissivity of 0.7 is portrayed in Figure V-28 in a similar manner as in Figure V-27. For the same thickness of conductor, the peak wall temperature is lowered 100°F .

In either case, the probable high densities which nature combines with high conductivity would make this technique extremely heavy.

If 0.7 inch of such a material were added to a 2.0 inch diameter shell for 8.0 inches the weight would be 17.9 lb, assuming silver to be such a material.

A 5-lb thrust unit was analyzed in a similar manner, results of which appear in Figure V-29. An actual design was utilized with reflects the realistic geometries involved.

Four variations were made:

Gas temperatures of 3500°F and 2800°F ; wall material of columbium, zirconium alloy, and a high axial conductor, all radiation cooled out the exit plane.

A contrasting size unit of 5 lb thrust was selected for comparison and a rational design, BAC Part No. 8348-470002, was studied. It is made of 99% CB/1% Z alloy and has a chamber configuration of unique shape, a diameter maximum between injector and nozzle.

Figure V-29. was analyzed as an axial conductor with internal radiation out the exit and insulation surrounding the exterior walls. The original columbium alloy was investigated first at a gas temperature of 3500°F and also 2800°F. Wall thicknesses were those of the part drawing. Adaptation of a 0.5 in. thick conductor of high conductivity resulted in a gross leveling of peak temperatures.

For small engines in the 5-lb thrust class, there is significant potential in the consideration of axial conduction.

e. Adiabatic Chamber

The adiabatic chamber is a most attractive candidate due to its simplicity, if the injector can be developed with lower heat flux or if higher temperature coatings are available. To evaluate the possibility of insulating a refractory wall chamber, analyses were conducted using a composite insulation system. Following an initial survey of "off the shelf" insulation materials, four materials were selected based on their temperature and thermal conductivity properties. Two designs were analyzed using three of the insulation materials for each design applied in concentric rings around the thrust chamber. The type, conductivity and temperature limits of these materials are listed in Table V-7. The conductivities are associated with the temperature limits shown in the table. Further discussion of the insulation materials is given in another section of this report.

Zirconia foam was initially selected obviously due to its high temperature capability, covered by "Dyna Quartz" and "MIN-K", in that respective order.

A two-dimensional steady state type heat balance was utilized which defined the thicknesses and outside wall temperatures portrayed in Figure V-30. The heat balance was:

TABLE V-7
INSULATORS FOR COMPOSITE INSTALLATION ON ACTUAL THRUST CHAMBER

<u>Design</u>	<u>Zone</u>	<u>Material</u>	<u>Temperature Limits (°F)</u>	<u>Thermal Conductivity B in. ft²hr° F</u>
I	A. Adjacent to thrust chamber inner metallic shell	Zirconia Foam	2700-3500	5.0
	B. Outside of (A)	"Dyna Quartz"	2200-2700	0.70
	C. Outside of (B)	"MIN-K"	400-2200	0.40
II	A. Adjacent to thrust chamber inner metallic shell	Graphite Fibers	2700-3500	0.75
	B. Outside of (A)	"Dyna Quartz"	2200-2700	0.70
	C. Outside of (B)	"MIN-K"	400-2200	0.40
Chamber diameter = 2.0 inches				
Thrust = 100 lb				

$$R_1 h_g \left[T_g - T_1 \right] = \frac{K_1}{\ln R_2} \left[T_1 - T_2 \right] = \frac{K_2}{R_2 \ln R_3} \left[T_2 - T_3 \right]$$

$$= \frac{K_3}{\ln \frac{R_4}{R_3}} \left[T_3 - T_4 \right] = R_4 \left[\sigma \epsilon (T_4^4) \right]$$

with

- R = material radius, in.
- $h_g = 0.0003 \text{ BTU/in.}^2 \text{ sec}^\circ\text{F}$
- $\epsilon = 0.7$ outside surface
- $T_g = 3500^\circ\text{F}$
- $T_2 = 2700^\circ\text{F}$
- $T_3 = 2200^\circ\text{F}$
- $T_4 = \text{variable}$

The first design with the hot side insulator of zirconia foam was investigated initially as it has the fortunate capability of high temperature oxidation resistance. The importance of this is that no inert atmosphere would be required for sea level testing and simplified test hardware would be an advantage. Figure V-30 indicates that a 9.8 inch diameter chamber would be required if a full 3500°F driving temperature was used. A reduction of 3500°F to 3000°F would permit an insulation package to be well within the size limitation and still have a 400°F outer wall.

Graphite fibers that exhibit a superior conductivity would meet the size limitation easily with 3500°F gas but requires an inert gas cannistor for seal level testing, Figure V-31.

f. Fuel-Rich Gas Regeneratively Cooled Chamber

This concept was investigated using propellant combinations other than that selected for this program. This concept appears feasible and certainly has merit for future consideration.

A Rocket Cooling System may be designed to include a gaseous coolant that is unhampered by throttling or chemical decomposition. This is especially applicable to the low thrust units with a high surface area to chamber volume ratio. It is acknowledged that a gas has approximately 1/4 cooling heat transfer coefficient of a liquid but, if the wall material has sufficient high temperature capability, a passage size may be found compatible with the coolant flow available and its source temperature. A description of the cooling system and method of analysis follows.

Fuel-rich gases are produced external to the main combustion chamber as per Figure V-32a, they pass along the walls as a coolant and are admitted to the main chamber at the injector end. These gases are then burned with the oxidizer. Heat transfer analysis is conducted by heat balance descriptive of the modes and amounts of heating that can be identifiable.

Consider the analytical mode as portrayed in Figure V-32b. The inner and outer walls are concentric to the thrust chamber axis with the coolant flowing between inner and outer walls. The outer wall temperature (T_{w2}) and inner wall temperature (T_{w1}) and coolant bulk temperature (T_L) are identifiable with the driving gas temperature (T_R). The system is a low heat transfer rate system so that the thermal gradient across the wall(s) may be considered insignificant. Also the heating source of the free stream gas was assumed to be infinite.

A steady state transferral of heat occurs with energy transferred from the free stream to the inner wall by forced convection. Some of the thermal energy of the inner wall is transferred to the outer wall by radiation and some by forced convection to the coolant. The coolant absorbs sensible heat from both inner and outer walls. With these statements we may put forth two expressions for heat flux.

Considering first the isolates system enclosing the inner wall (1), we may write

$$\dot{q}_1 = \dot{q}_2 + \dot{q}_3 \quad (1)$$

and for the system, across the outer wall (2)

$$\dot{q}_4 = \dot{q}_5 \quad (2)$$

- $\dot{q}_1$ = the total driving energy of the free stream
 $\dot{q}_2$ = energy expelled from wall (1) absorbed wall (2) by radiation
 $\dot{q}_3$ = energy lost from wall (2) to the coolant gas
 $\dot{q}_4$ = radiant energy transferred from wall (1) to wall (2)
 $\dot{q}_5$ = heat extracted from wall (2) by the coolant gas

Equations (1) and (2) may be written

$$h_g (T_r - T_{w_1}) = F_{12} \sigma \epsilon_{in} (T_{w_1}^4 - T_{w_2}^4) + h_{L_1} (T_{w_1} - T_L) \quad (3)$$

$$F_{12} \sigma \epsilon_{in} (T_{w_1}^4 - T_{w_2}^4) = h_{L_2} (T_{w_2} - T_L) \quad (4)$$

Assuming no end effects, the inner and outer walls have a common view factor (F_{ij}). Hence $F_{12} = F_{21} = 1$. Since h_L is only slightly sensitive to wall temperature, $h_{L_1} = h_{L_2} = h_L$.

Rearranging Equations (3) and (4)

$$T_{w_1}^4 = \left[T_{w_2}^4 (-\sigma \epsilon_{in} + \sigma \epsilon_{ex}) - h_L (T_L - T_{w_2}) \right] \frac{1}{\sigma \epsilon_{in}} \quad (5)$$

$$T_{w_2}^4 = T_{w_1}^4 - \frac{1}{\sigma \epsilon_{in}} \left[h_g (T_r - T_{w_1}) h_L (T_{w_1} - T_L) \right] \quad (6)$$

The amount of energy absorbed by the coolant gas may be equated to the convective loadings on walls as

$$\dot{q}_G = \dot{w} C_p \Delta T \cong \left[\bar{h}_L (T_{w_2} - T_L) + \bar{h}_L (T_{w_1} - T_L) \right]_j A_{c_{j+1}} \quad (7)$$

From Equation (7), the temperature rise of the coolant gas over an incremental area is added to the bulk temperature entering the element (T_{L_j}).

$$T_{L_{j+1}} \cong T_{L_j} + \frac{\left[h_L (T_{w_1} + T_{w_2} - 2T_L) \right]_j A_{c_{j+1}}}{\dot{w} C_p} \quad (8)$$

where the surface area over an element is

$$j A_{c_{j+1}} = 2 (R_j + R_{j+1}) \pi \Delta L \quad (9)$$

It should be noted that the right hand side of Equation (7) is approximately equal to the heat absorbed by the gas; since the wall temperature T_{w1} and T_{w2} are for one unique point on the walls. But as long as the calculation procedure is performed over small incremental distance, no significant error would be introduced.

In addition, the adiabatic wall or recovery temperature is to be found for the area velocity core gas stream expressed as

$$T_R = T_o \left[\frac{1 + 0.89 \frac{\gamma - 1}{2} M^2}{1 + \frac{\gamma - 1}{2} M^2} \right] \quad (10)$$

0.89 = turbulent recovery factor for air.

The Mach number is found from the local known area ratio as

$$\left(\frac{R}{R^x} \right)^2 = \frac{1}{M} \left[\frac{2}{\gamma + 1} \left(1 + \left[\frac{\gamma - 1}{2} M^2 \right] \right) \right]^{\frac{\gamma + 1}{2(\gamma - 1)}} \quad (11)$$

The simultaneous solution of Equations (5) and (6) yields solution of the wall temperatures T_{w1} and T_{w2} . These in turn are used to find the coolant temperature rise by means of Equation (7). Successive solutions to axial increments of thrust chambers have been programmed for the Bell IBM 7090 computer and are identified as program 3217.

This program was used in determining the wall temperature distribution for the following propellant combinations.

H_2O_2	- N_2H_4
F_2	- N_2H_4
OF_2	- N_2H_4
Cmpd "A"	- N_2H_4
CTF	- N_2H_4

when used in a 100 lb thrust unit at 80 psia chamber pressure. Results of two of these solutions for inlet temperatures of 2060°R appear in Figures V-33 and V-34. A plot of maximum wall temperatures of the inner liner versus core driving temperature for all the propellant combinations appears in Figure V-35. Two inlet coolant temperatures were investigated, 1460°R and 2060°R.

The increase in wall temperature was observed to be linear with driving temperature but not a "one to one" increase; a 1000°R increase in core gas temperature results in only a 500°R increase in wall temperature. At this time the production of a low temperature gas in a gas generator is open to a certain amount of conjecture, so two temperature limits were assumed. A 600°R difference between these results in only a 250°R difference in maximum wall temperature.

This study was conducted with only one variation of coolant film coefficient. If the coefficient was increased, the most immediate result would be seen in the coolant temperature rise, and secondly in the wall temperature.

Figure V-36 presents a preliminary sketch of the gas regeneratively cooled chamber concept with a bipropellant valve installed.

Another common characteristic of the cases shown is that through the throat region the wall temperature distributions diverge, and on the chamber and divergent nozzle sections the distributions either remain parallel or converge. This may be explained by considering two cases. As limits of all convective heat transfer and all radiation heat transfer

Case I $h_g = h_L \longrightarrow \infty$

$$\frac{\dot{q}_{\text{rad}}}{\dot{q}_{\text{conv.}}} \longrightarrow 0$$

From a basic equation

$$h_L (T_{w_2} - T_L) = \sigma \epsilon (T_{w_1}^4 - T_{w_2}^4)$$

$$(\text{assuming } \epsilon_{\text{ex}} \cong 0)$$

the right hand side approaches a small value with respect to h_L , to have an identity, $T_{w_2} = T_L$.

Investigation of the second equation yields

$$h_g (T_r - T_1) = \sigma \epsilon (T_{w_1}^4 - T_{w_2}^4) + h_L (T_{w_1} - T_L)$$

the radiation term may be neglected and

$$T_r - T_{w_1} \cong T_{w_1} - T_L$$

or

$$T_{w_1} \cong \frac{T_r + T_L}{2}$$

Therefore, as $H_g = h_i$ becomes large, throat region, $T_{w_2} - T_L$ and

$T_{w_1} \frac{T_r + T_L}{2}$; i.e., divergence. For the other limit we have

Case II $h_g = h_L \longrightarrow 0$

$$\frac{\dot{q}_{\text{rad}}}{\dot{q}_{\text{con}}} \longrightarrow \infty$$

Since $h_L (T_{w_2} - T_L) \longrightarrow 0$, $T_{w_1}^4 - T_{w_2}^4 \longrightarrow 0$, therefore $T_{w_1} = T_{w_2}$.

Also $h_g (T_r - T_{w_1}) = h_L (T_2 - T_L) + h_L (T_{w_1} - T_L)$, therefore

$$T_{w_1} = T_{w_2} \frac{T_r + 2 T_L}{3}, \text{ convergence of parallelism between } T_{w_1} \text{ and } T_{w_2}.$$

TERMS

$\dot{q}$	heat
ΔL	length in.
R	radius in.
R^x	radius throat in.
T_p	total temperature °R
T_r	recovery temperature °R
T_{w_1}	inner wall temperature °R
T_{w_2}	outer wall temperature °R
T_L	coolant gas temperature °R
σ	Stefan Boltzmann constant
ϵ	emissivity
M	Mach No.
γ	ratio of specific heats
C_p	specific heat BTU/in. ² °R
$\dot{w}$	flow rate lb/sec
h_L	cooled film coefficient BTU/in. ² sec °R
h_g	gas film coefficient BTU/in. ² sec °R
F_{ij}	radiation view factor
A_c	coolant area in. ²

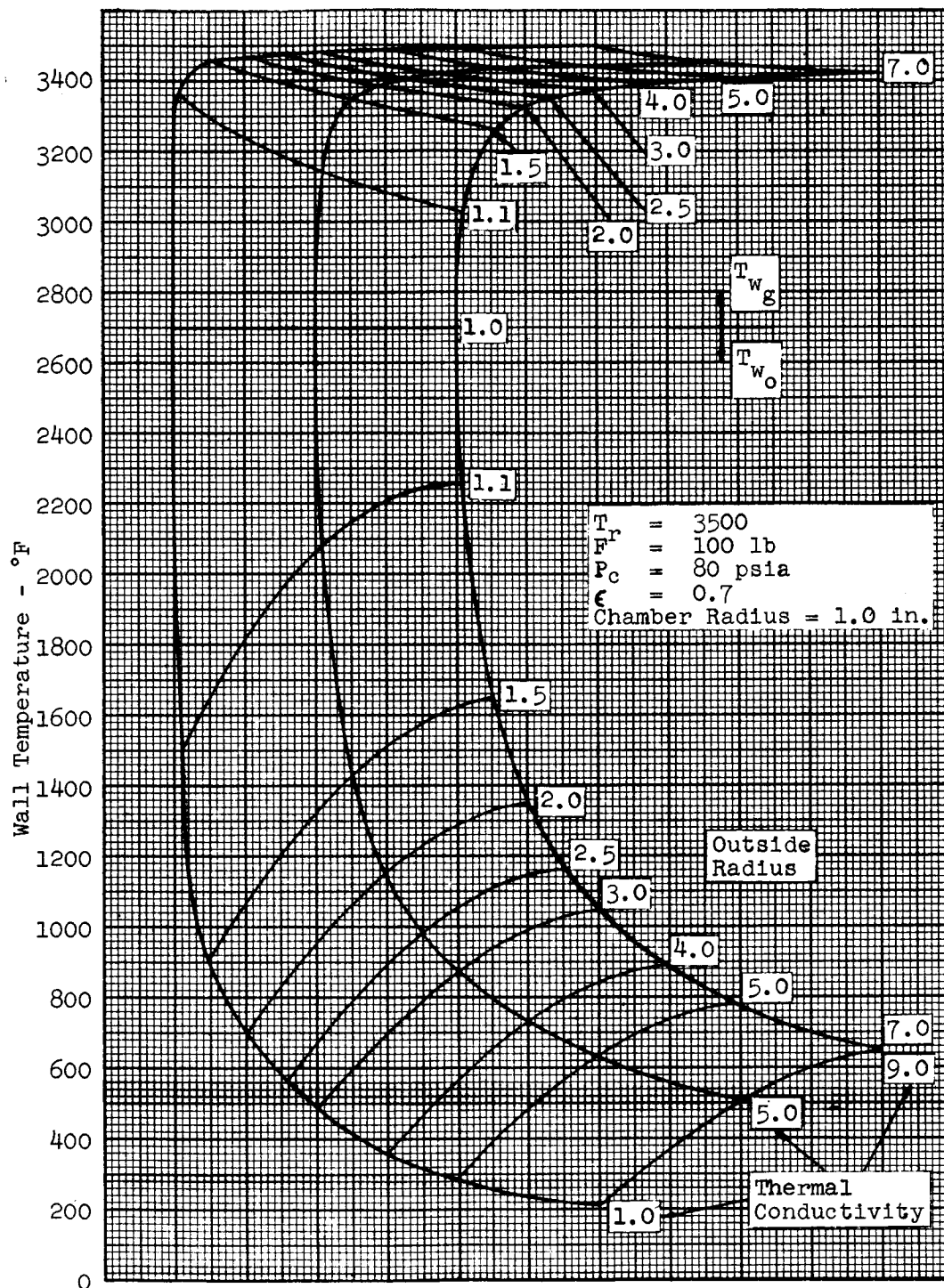


Figure V-5. Wall Temperature versus K and Outside Radius
Insulated Thrust Chamber

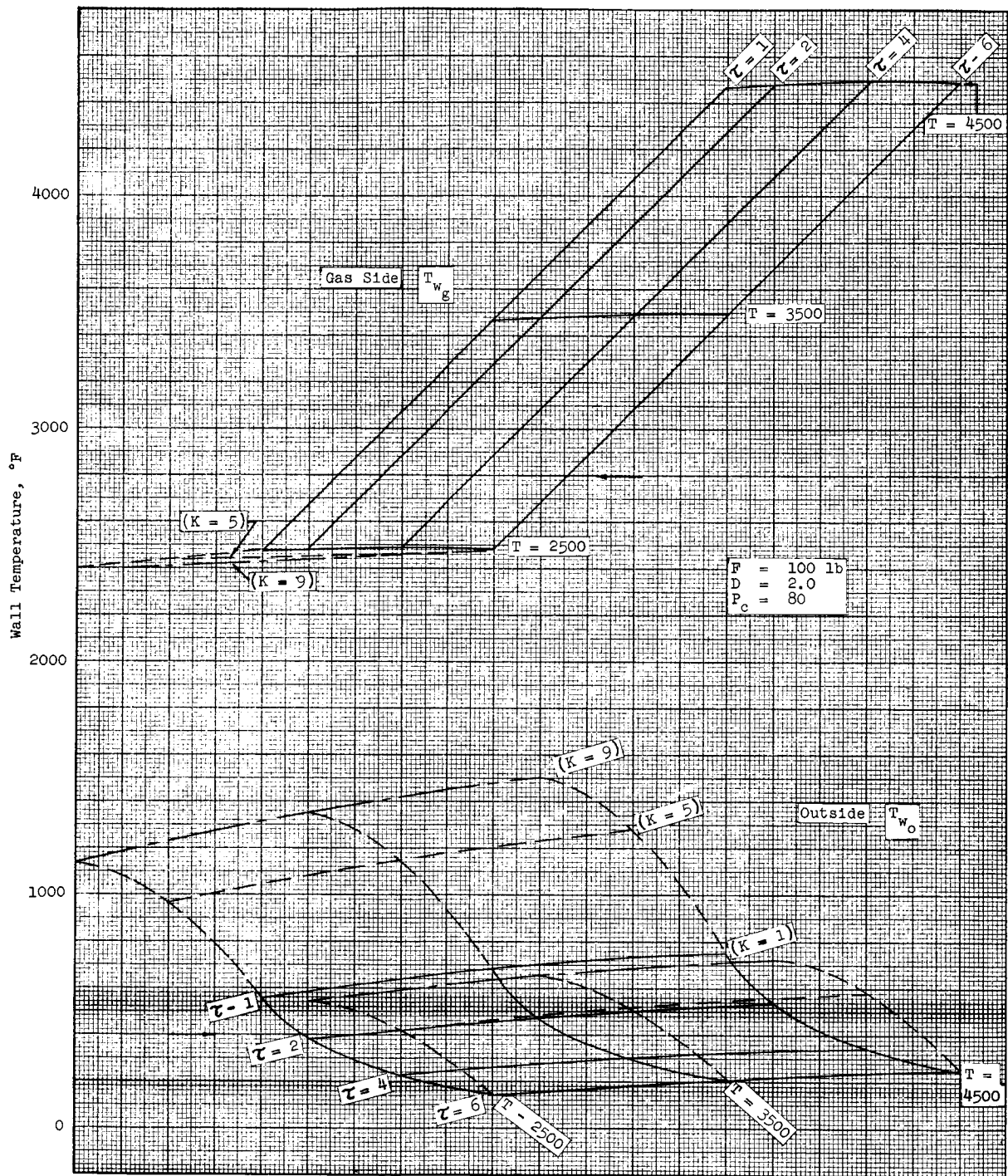


Figure V-6. Wall Temperature as a Function of T_G , T , and K - Insulated Thrust Chamber

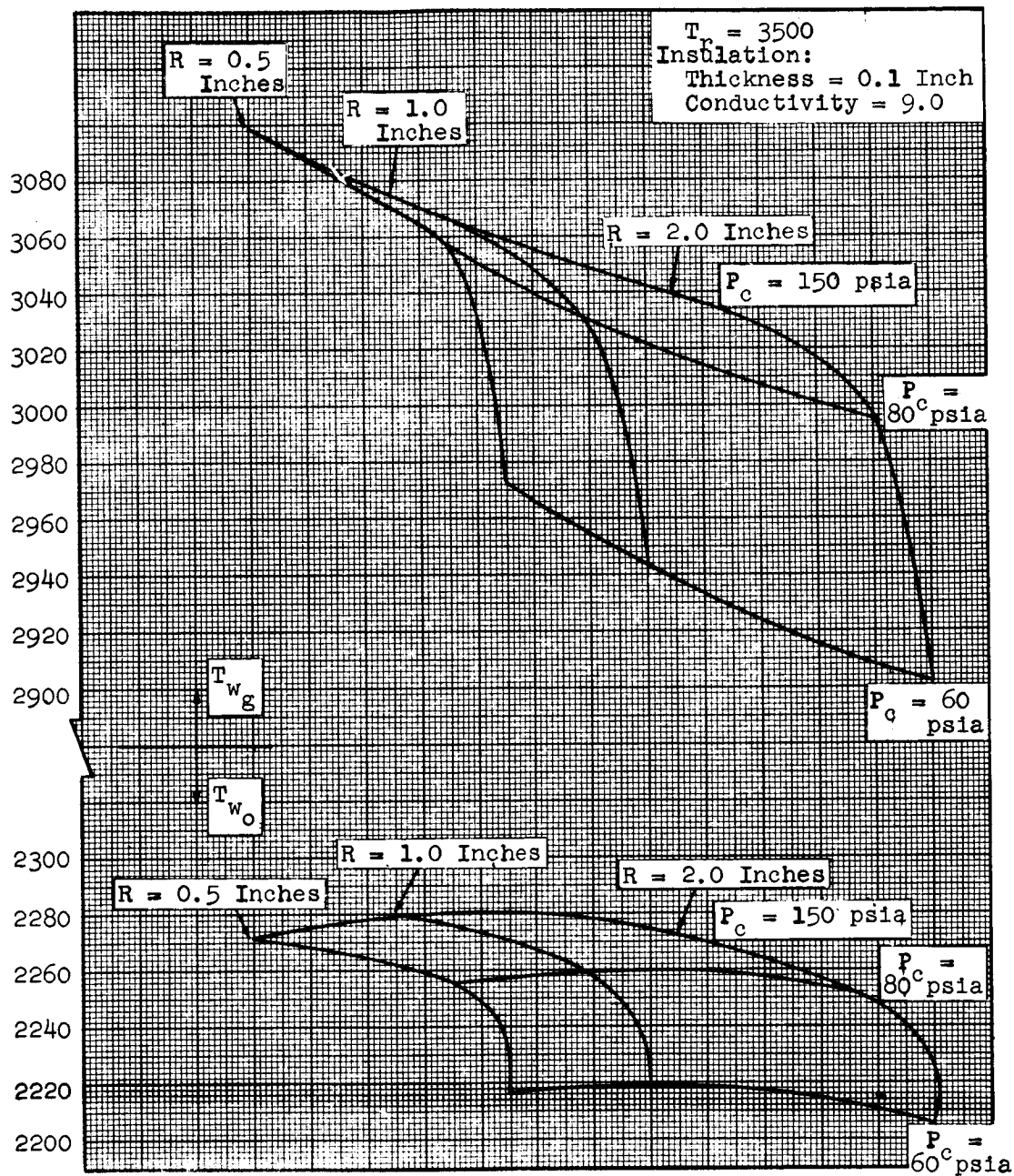


Figure V-7. Wall Temperature versus Chamber Pressure and Radius
Insulated Thrust Chamber

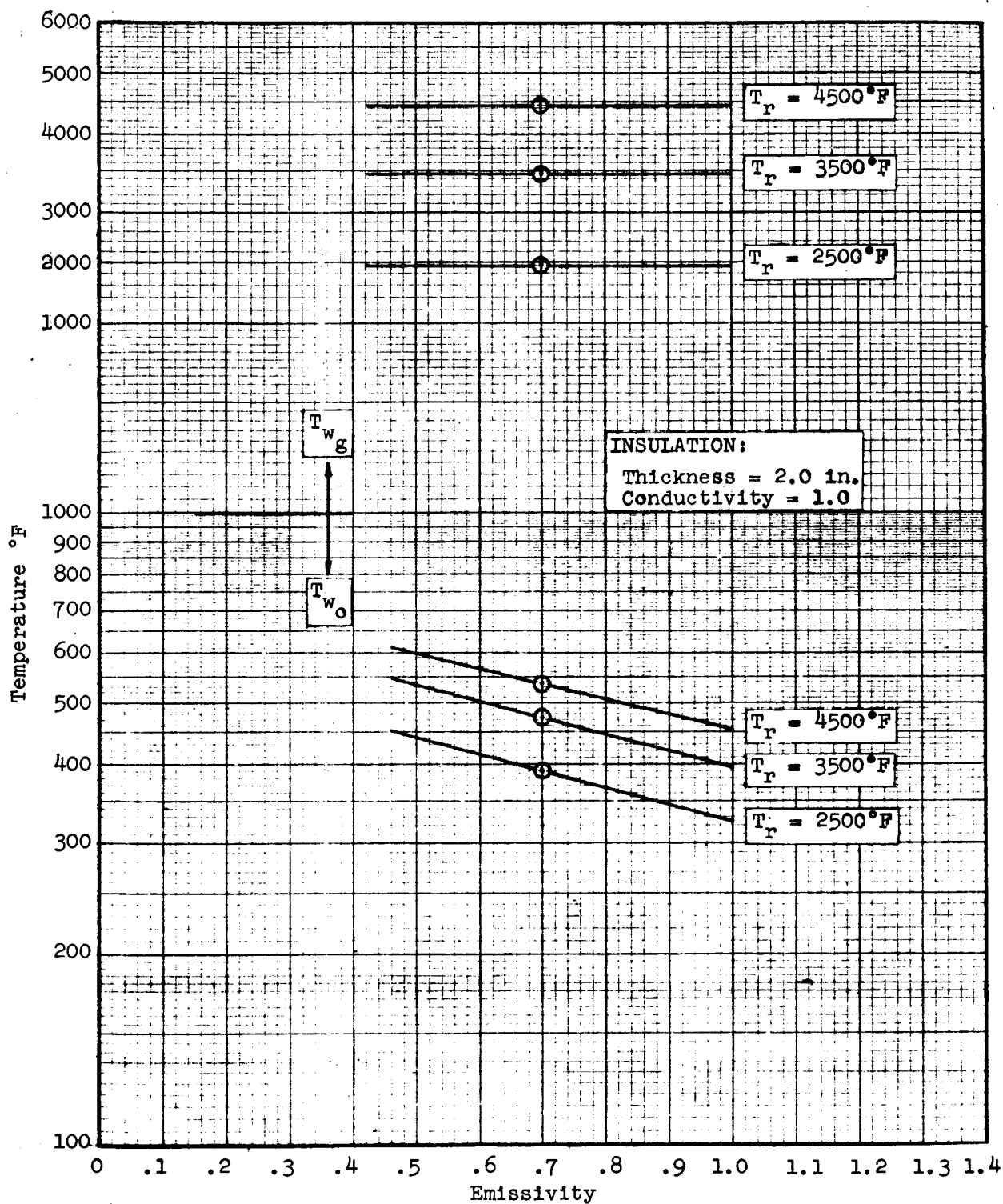


Figure V-9. Wall Temperature versus Gas Temperature and Emissivity Insulated Thrust Chamber

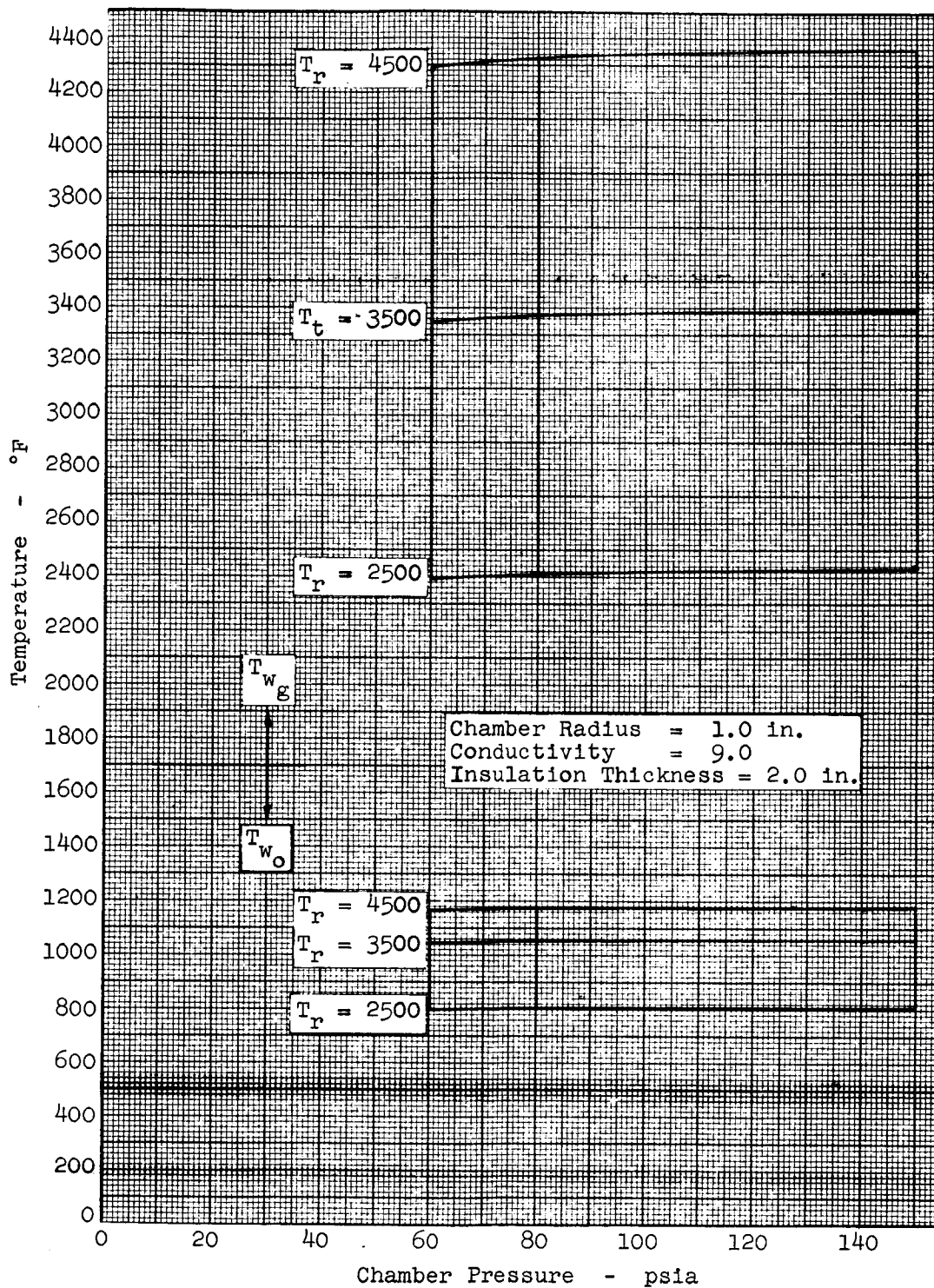


Figure V-8. Wall Temperature versus Gas Temperature and Chamber Pressure

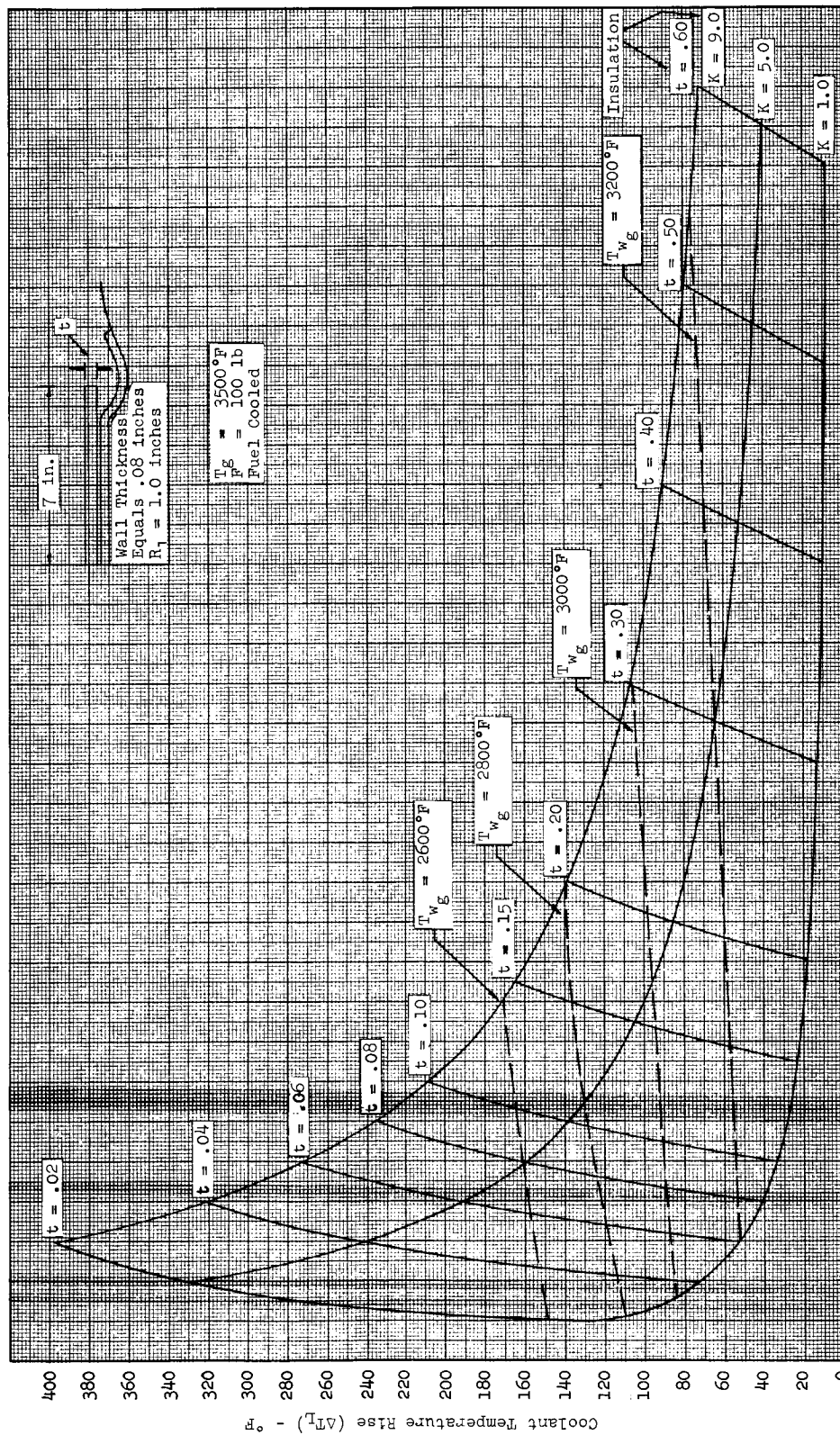


Figure V-10. Coolant Temperature Rise versus Insulation Thickness and Conductivity - Thermally Thick Thrust Chamber

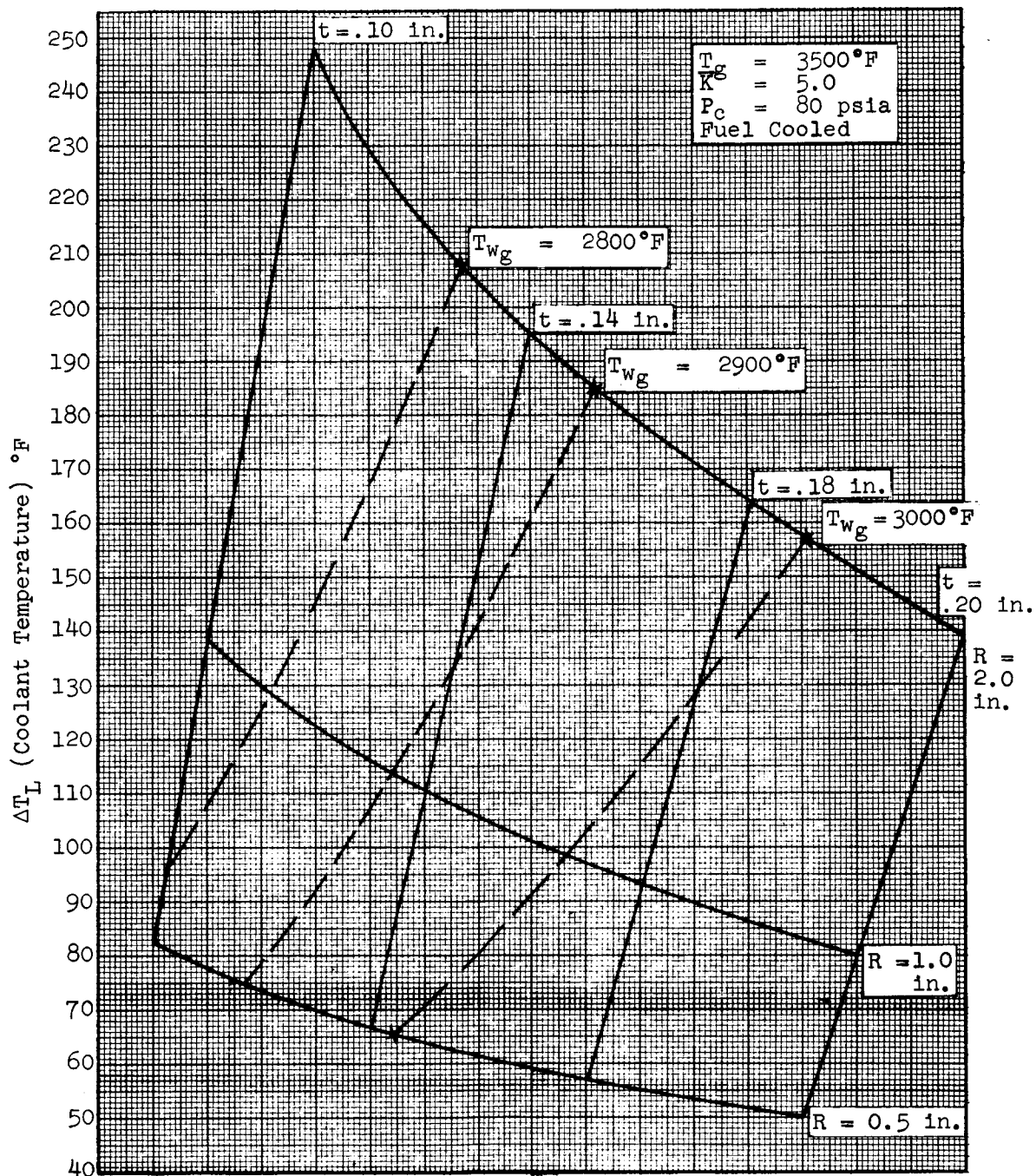


Figure V-11. Coolant Temperature Rise versus Thickness and Radius
Thermally Thick Thrust Chambers

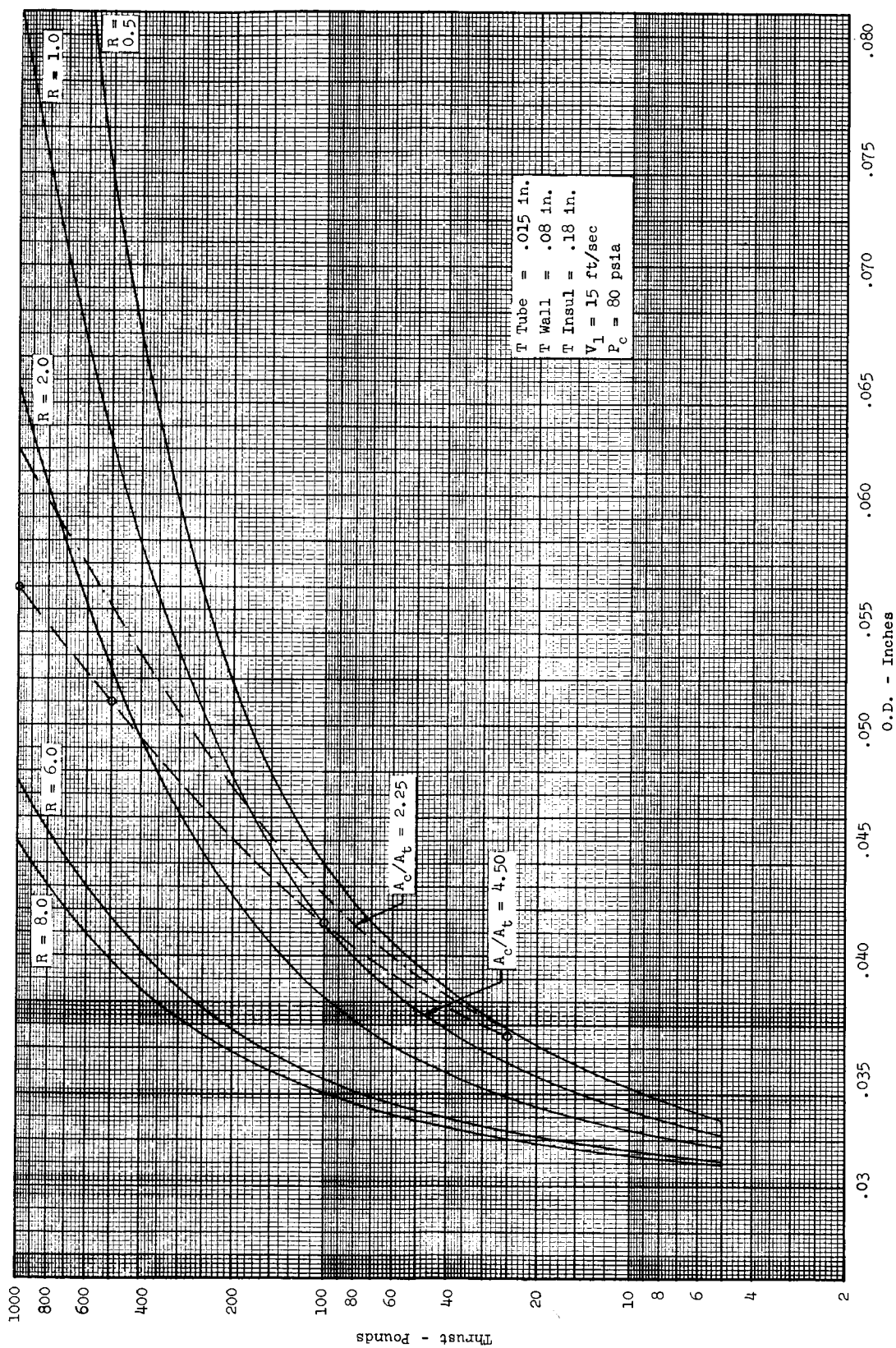
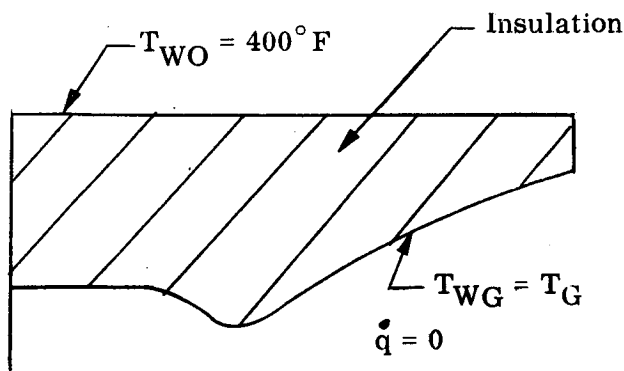
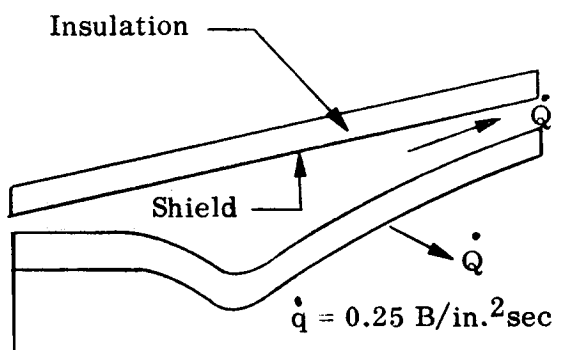


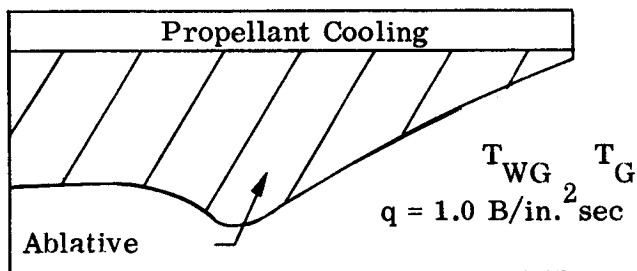
Figure V-12. Thrust versus Tube O.D. - Thermally Thick Thrust Chamber



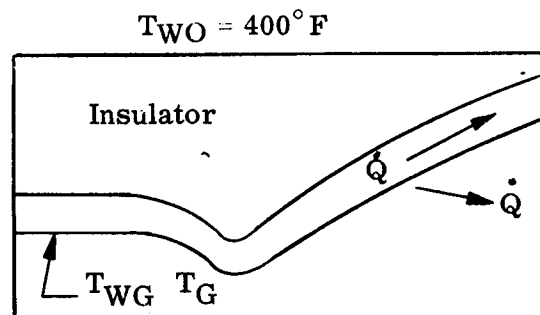
Adiabatic



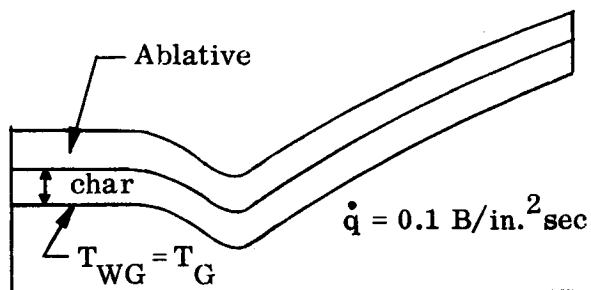
Radiation



Thermally Thick Regenerative

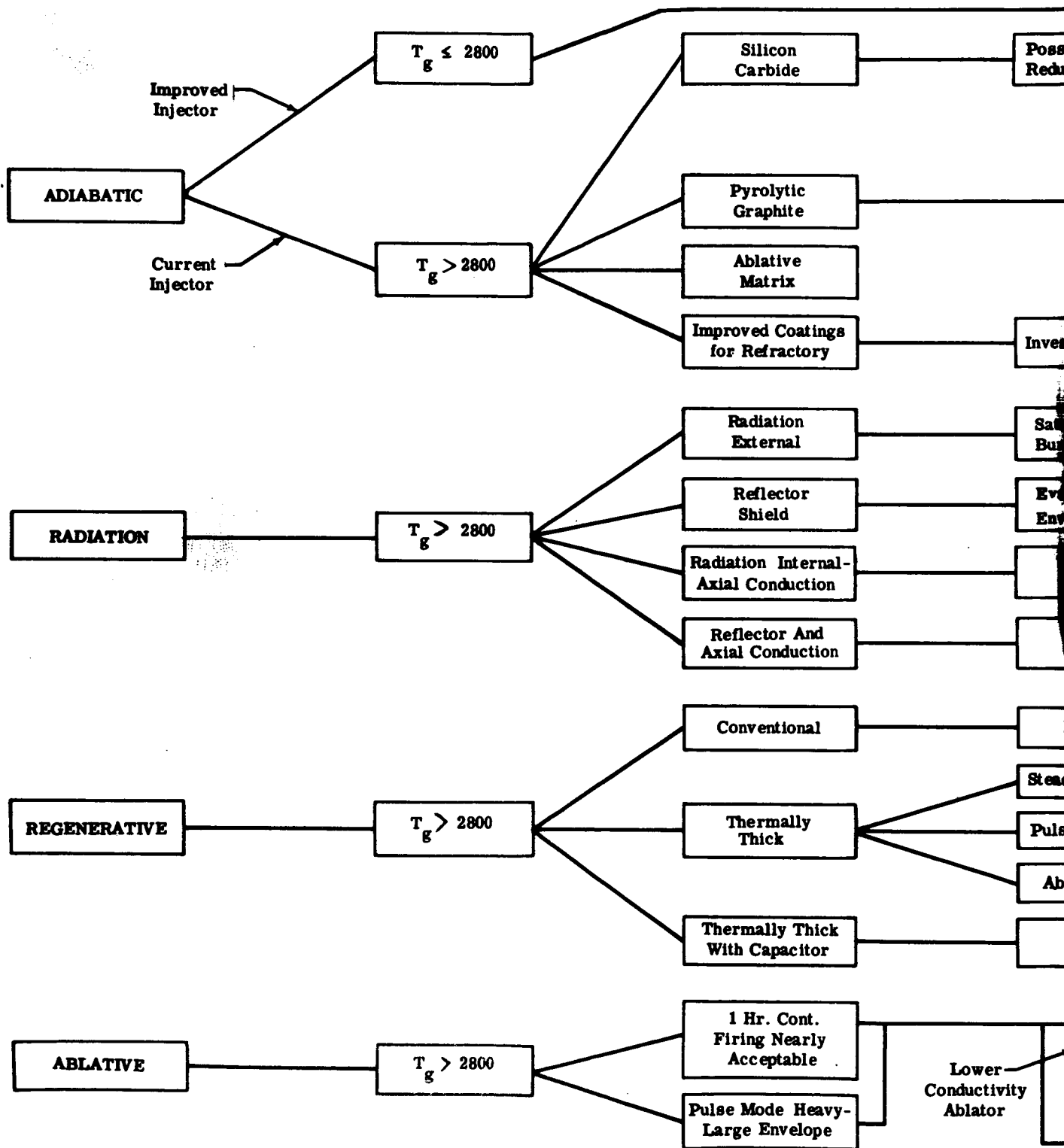


High Axial Conductor



Ablative

Figure V-13. Basic Thrust Chamber Concepts



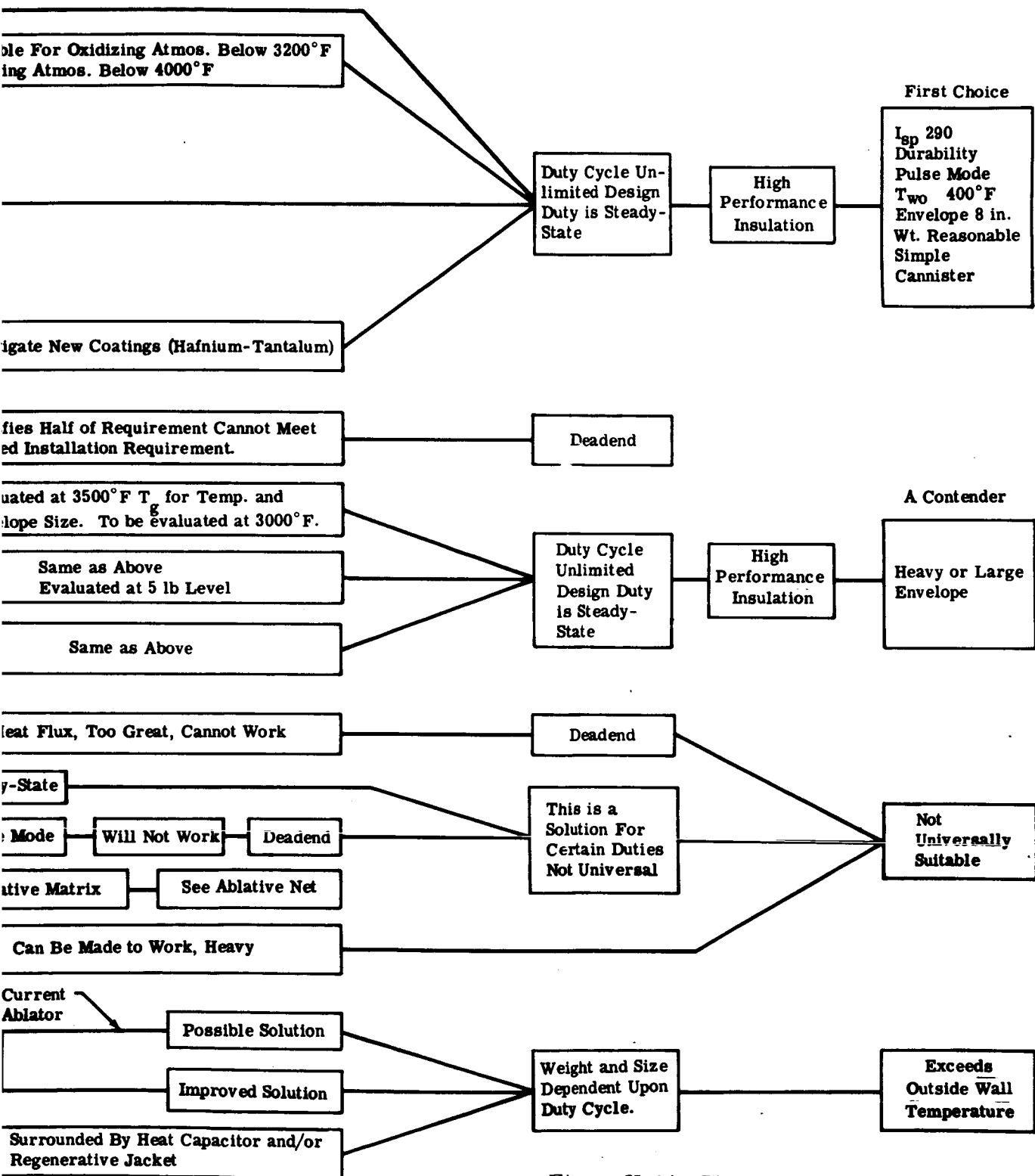


Figure V-14. Thrust Chamber Design Concepts

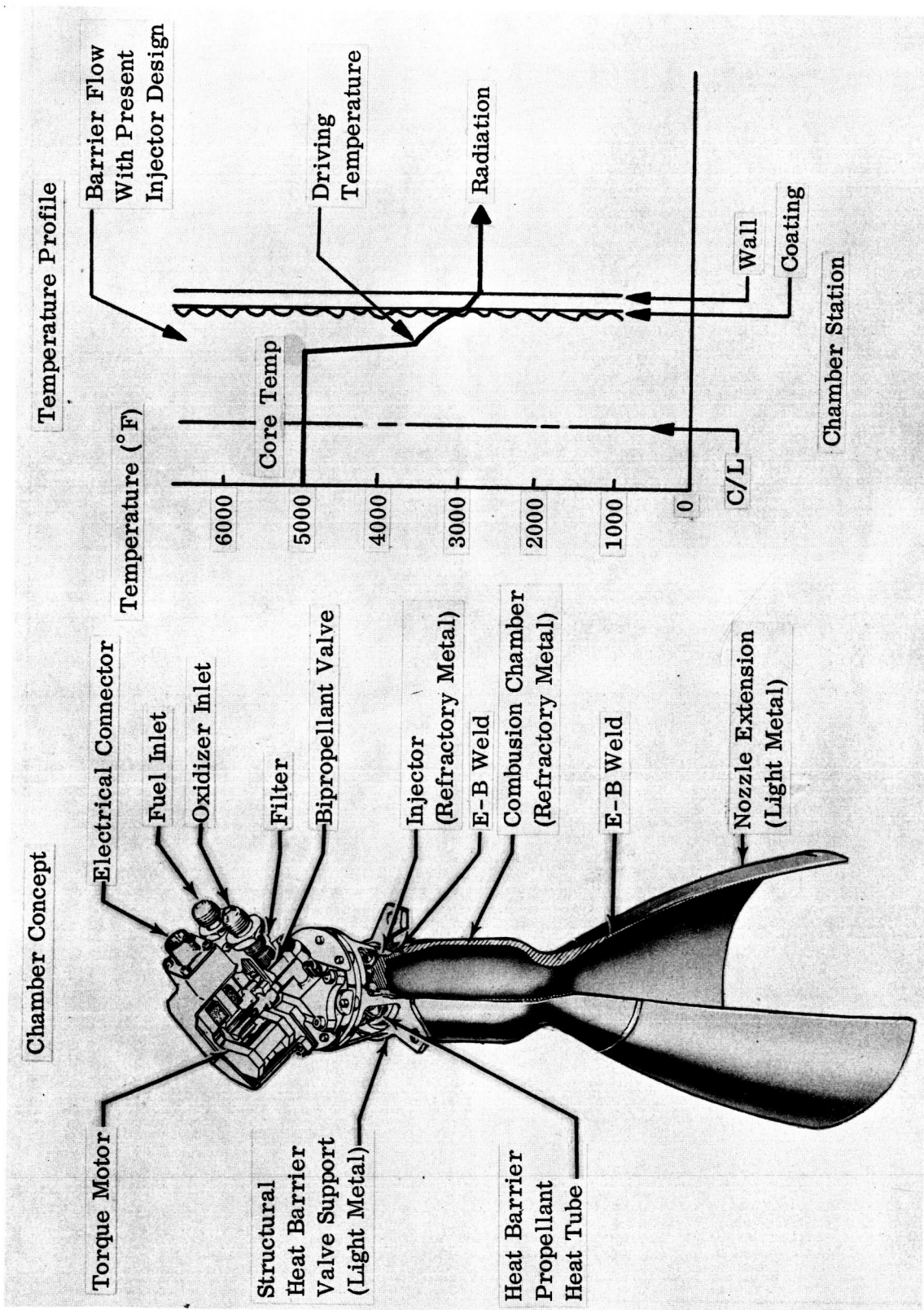


Figure V-15. Radiation Cooled Chamber

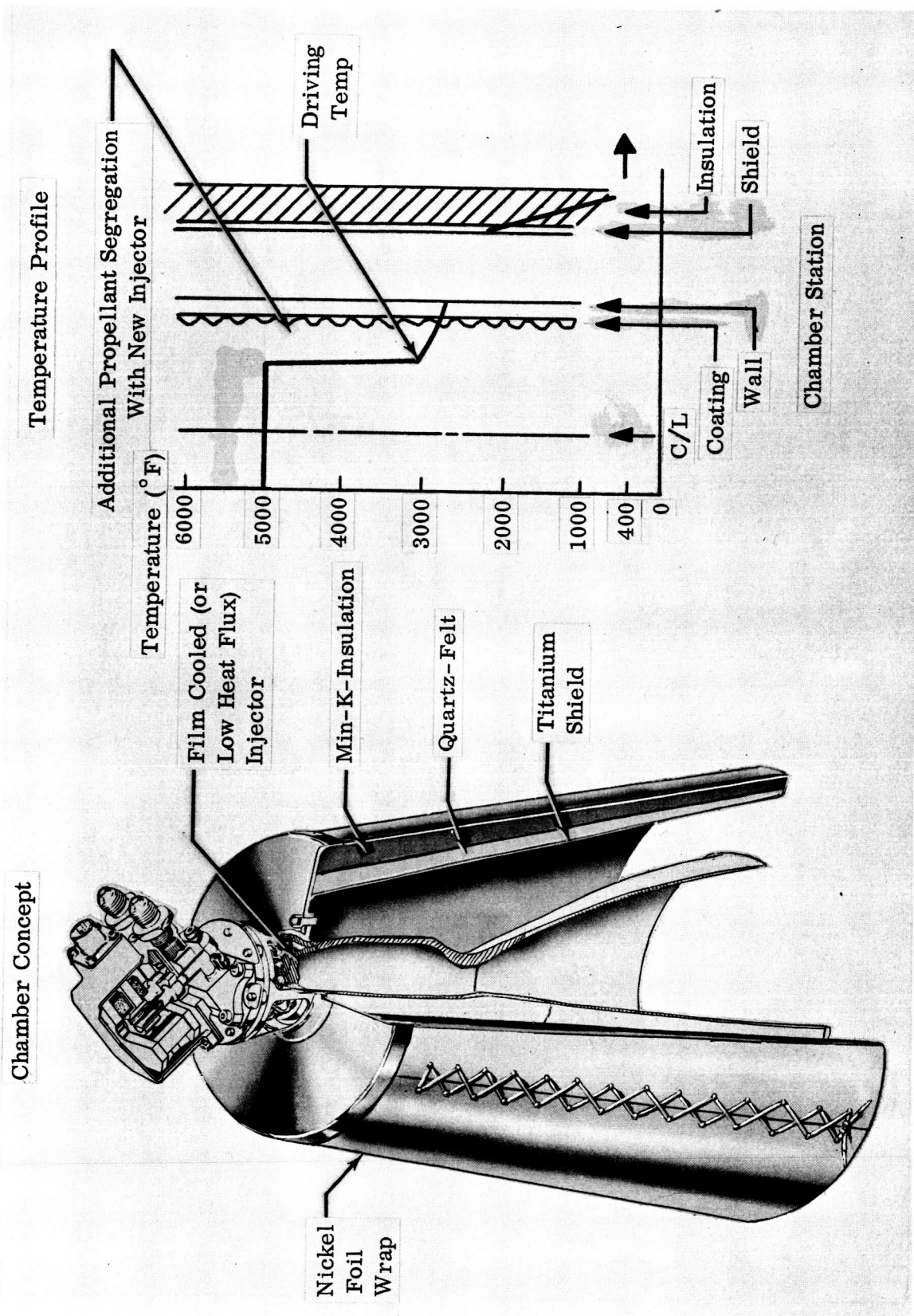


Figure V-16. Heat Shield Radiation Chamber

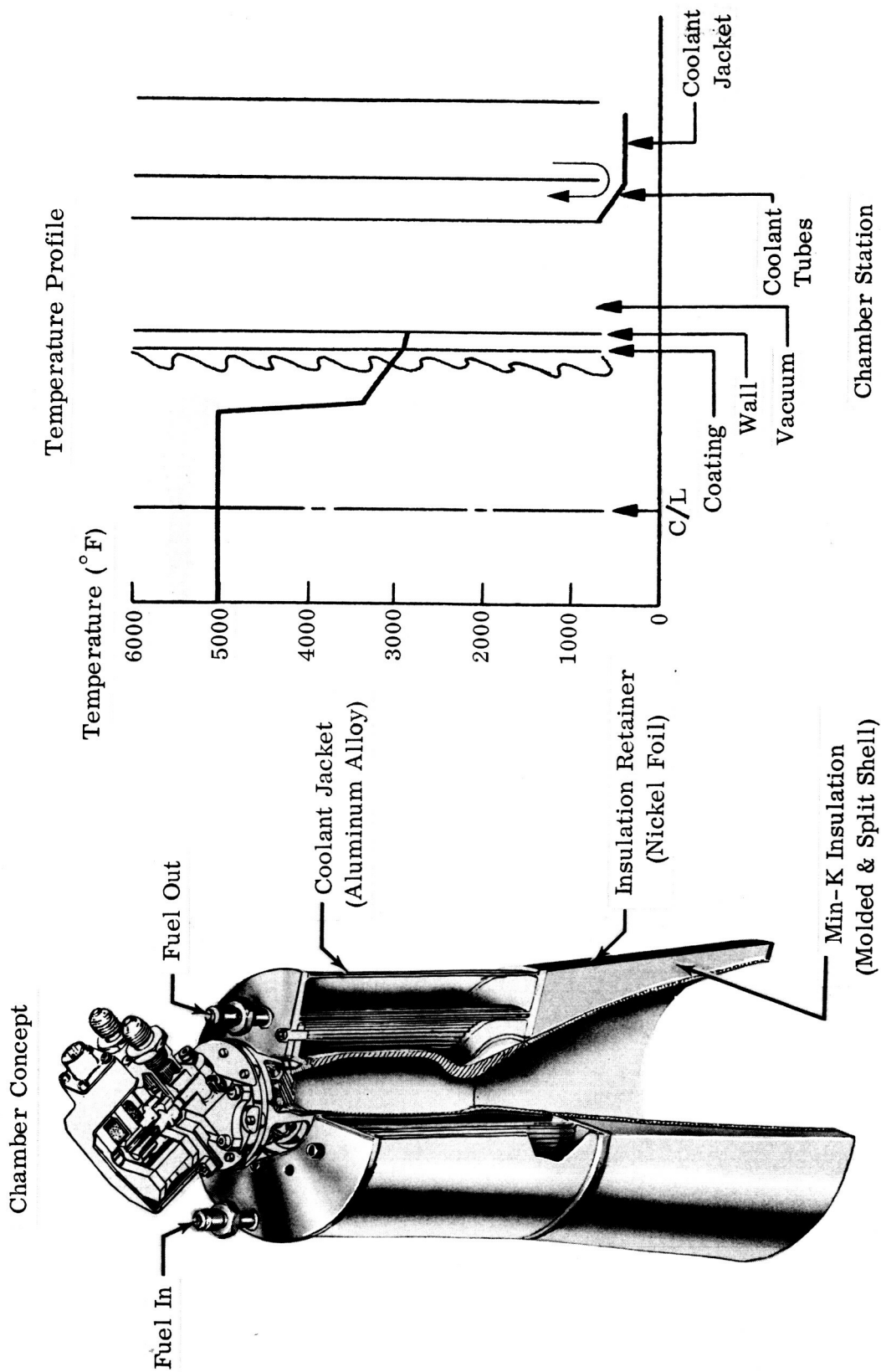


Figure V-17. Semi-Regenerative Chamber (Refractory)

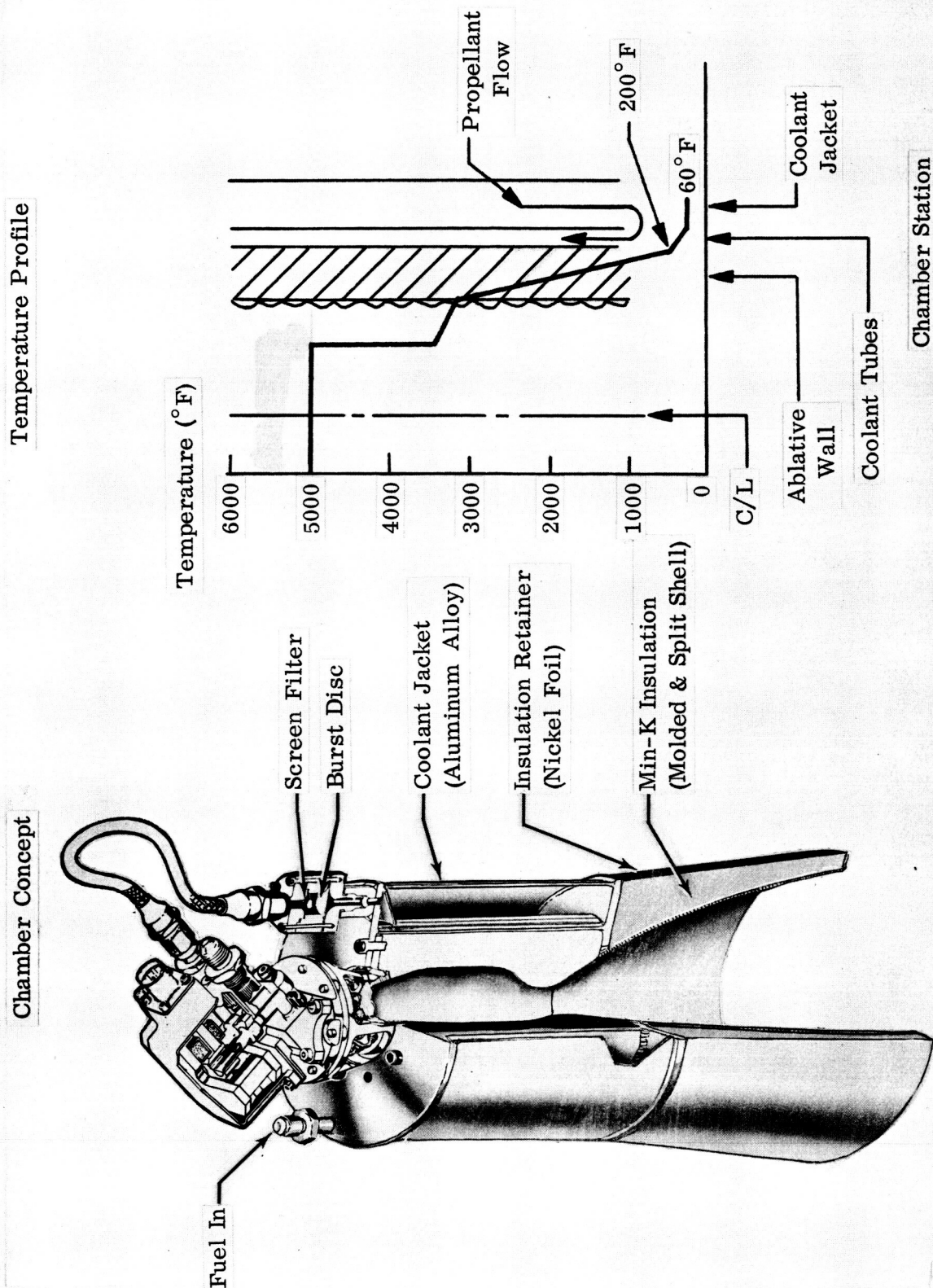
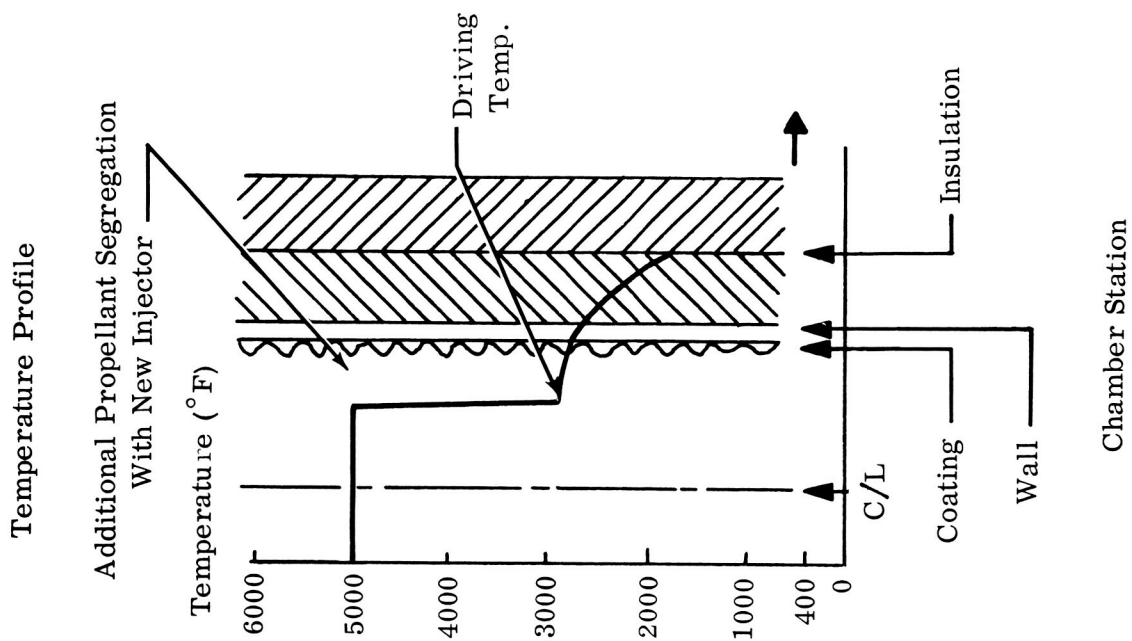


Figure V-18. Semi-Regenerative Chamber (Ablative)



Chamber Concept

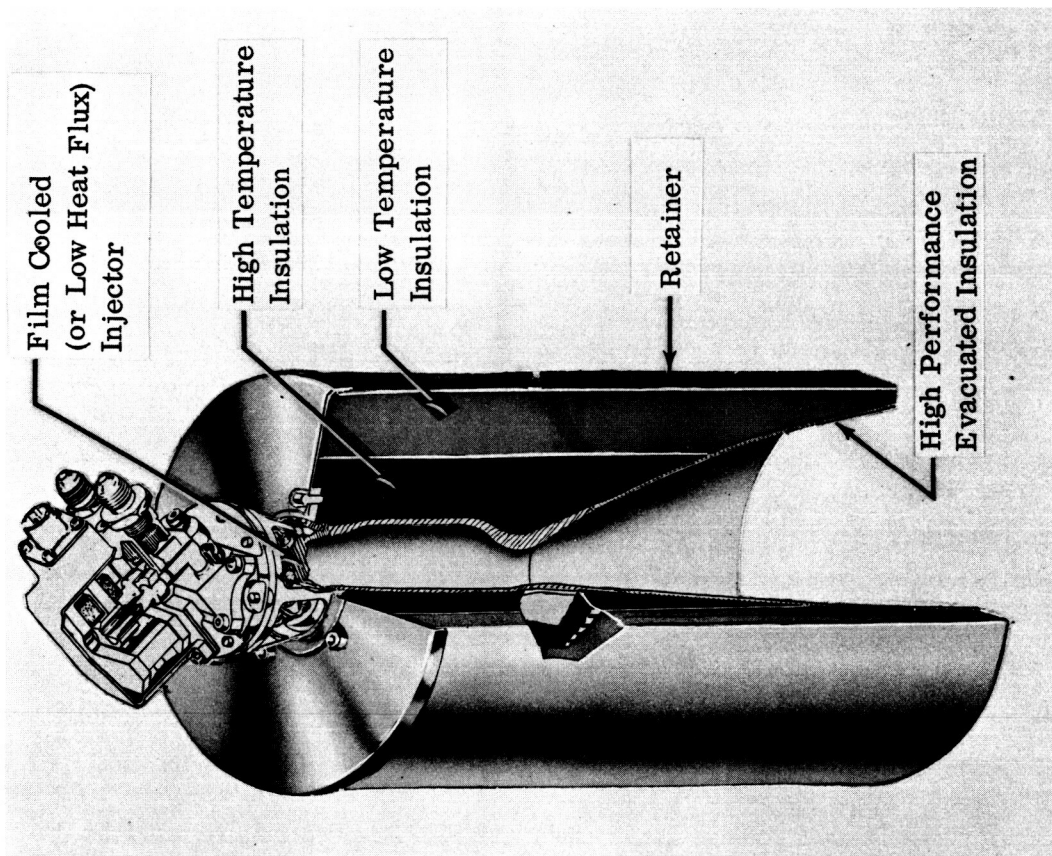


Figure V-19. Adiabatic Wall Chamber

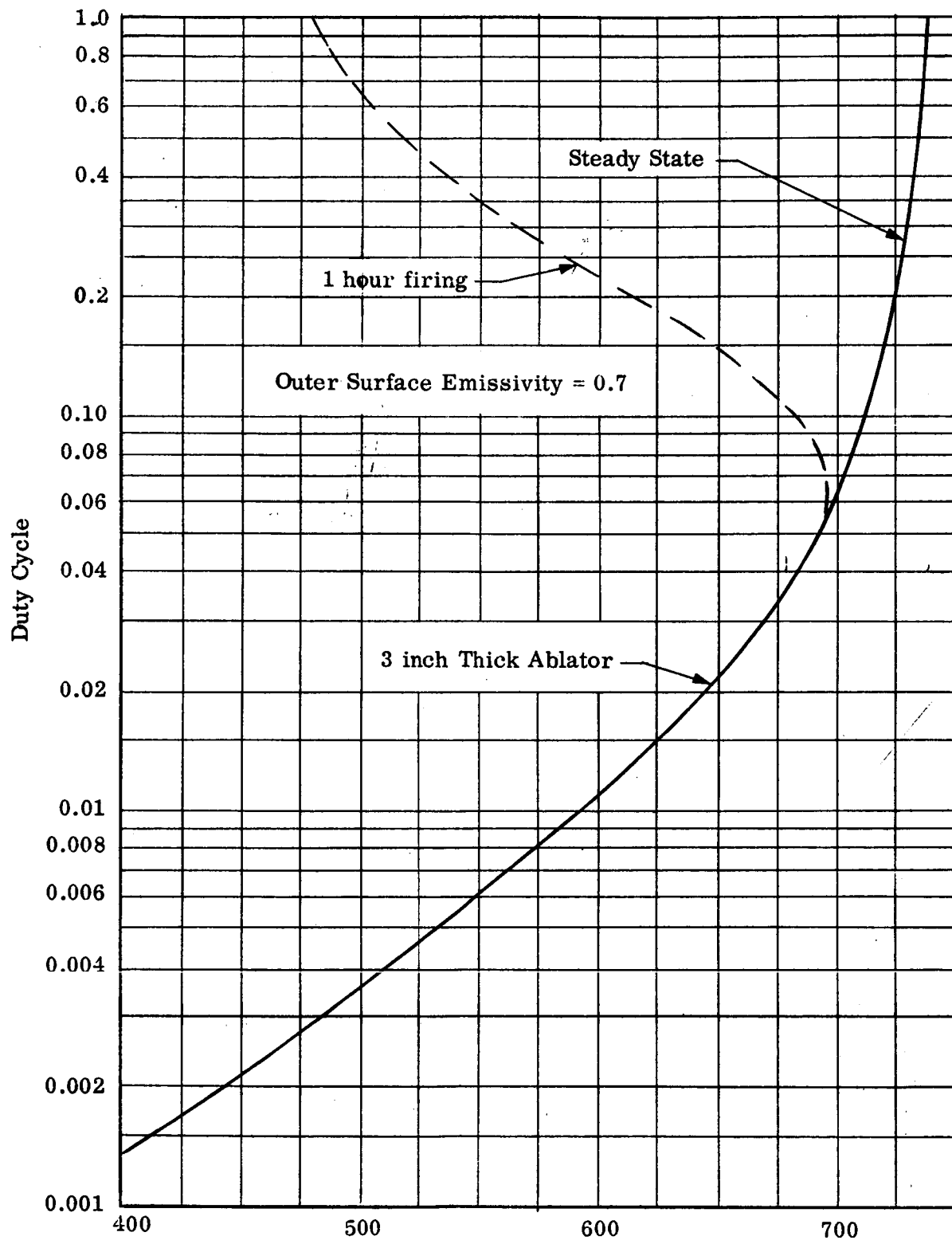


Figure V-20. Back Side Wall Temperature versus Duty Cycle for Ablative Chamber

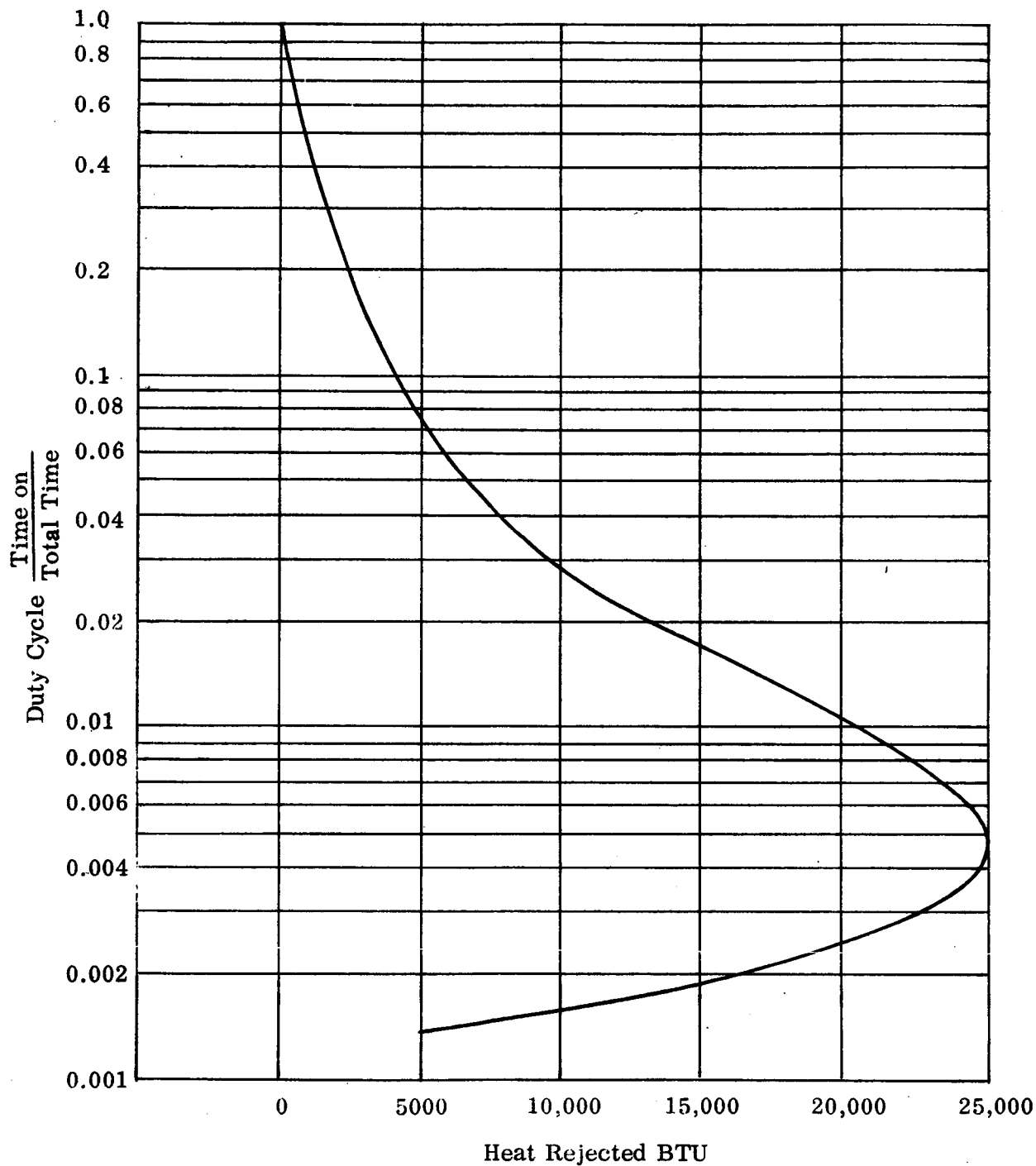


Figure V-21. Heat Output from 100-lb Ablative Engine versus Duty Cycle

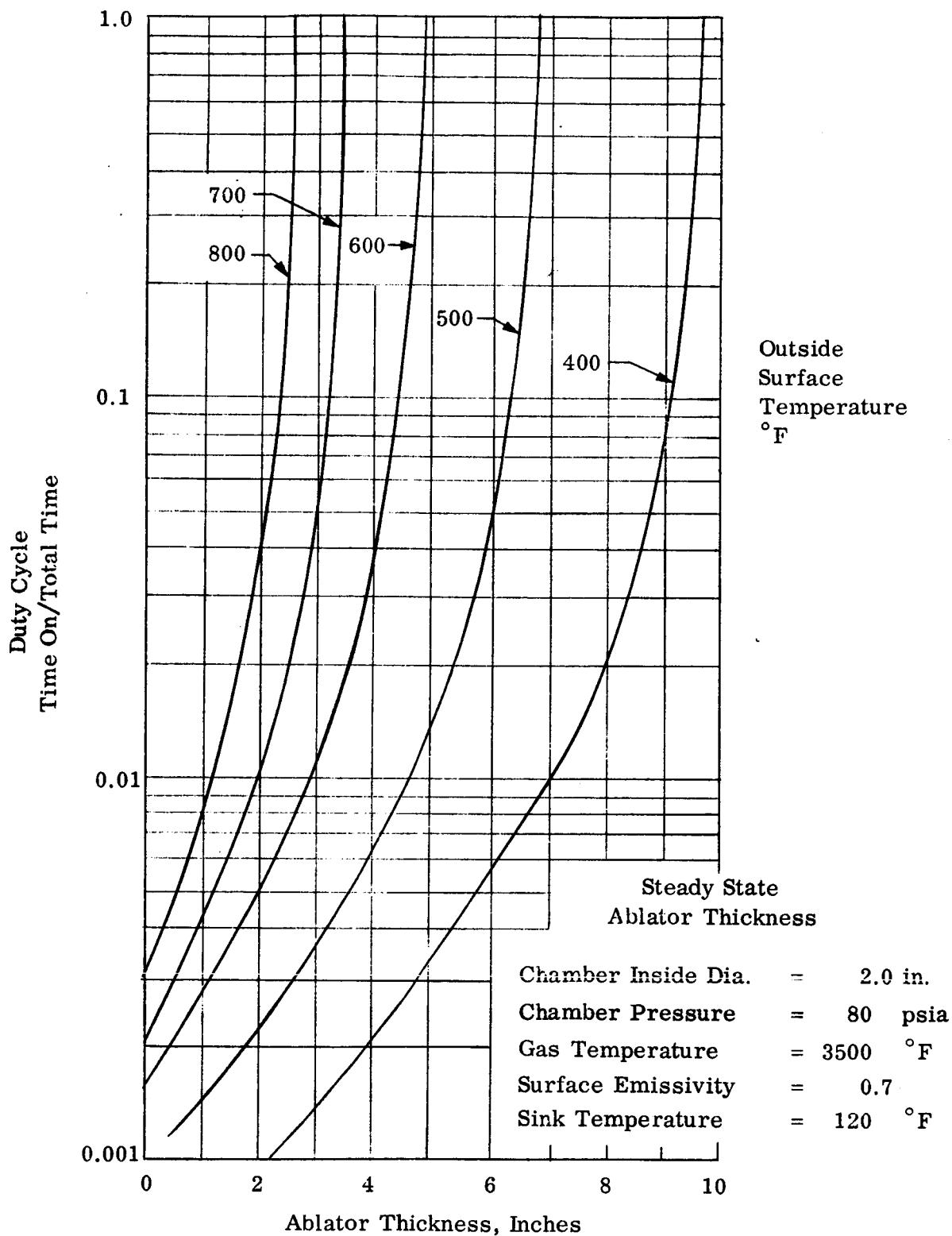


Figure V-22. Ablative Analysis - Pulse Operation

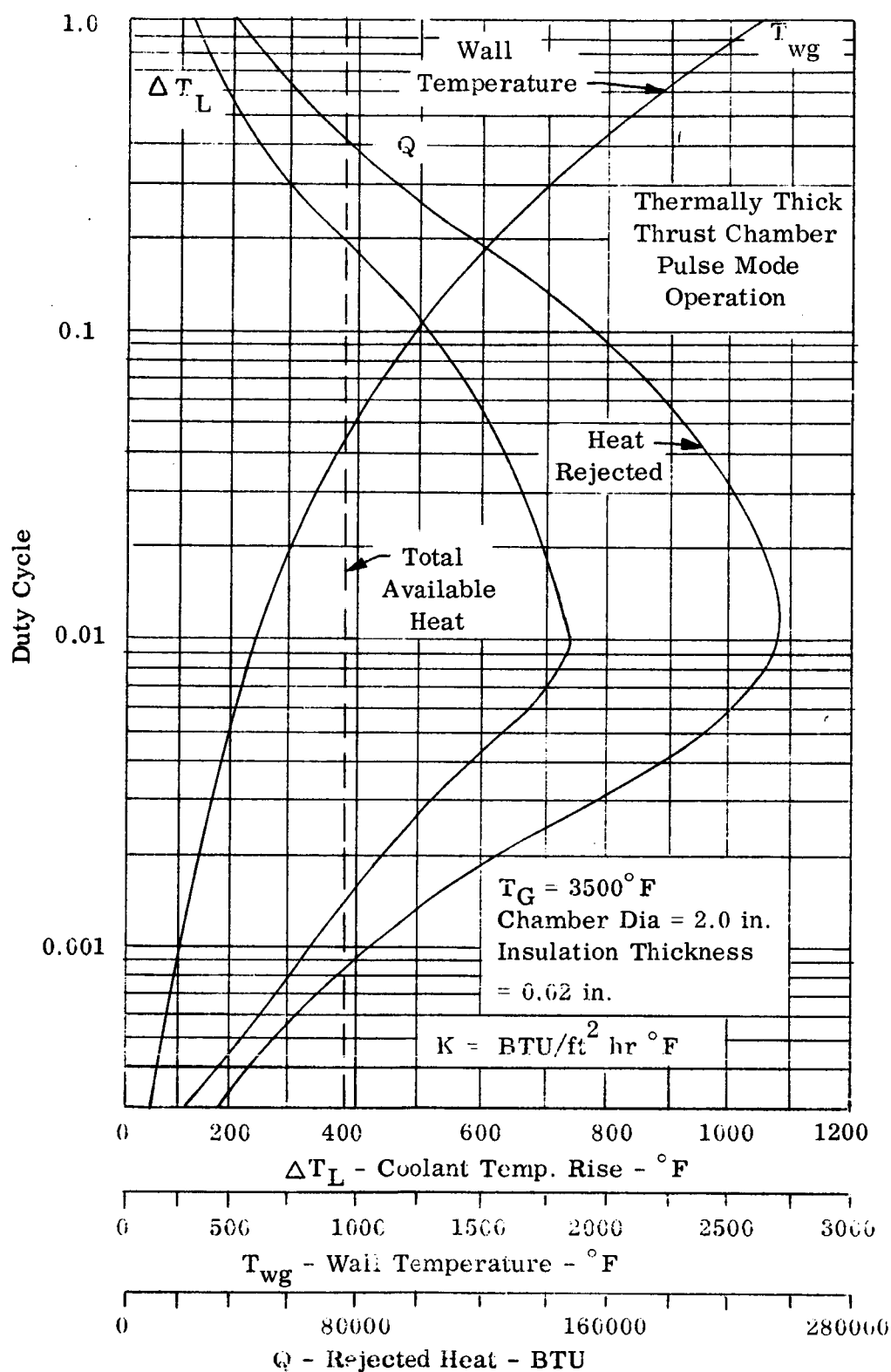


Figure V-23. Thermally Thick Thrust Chamber

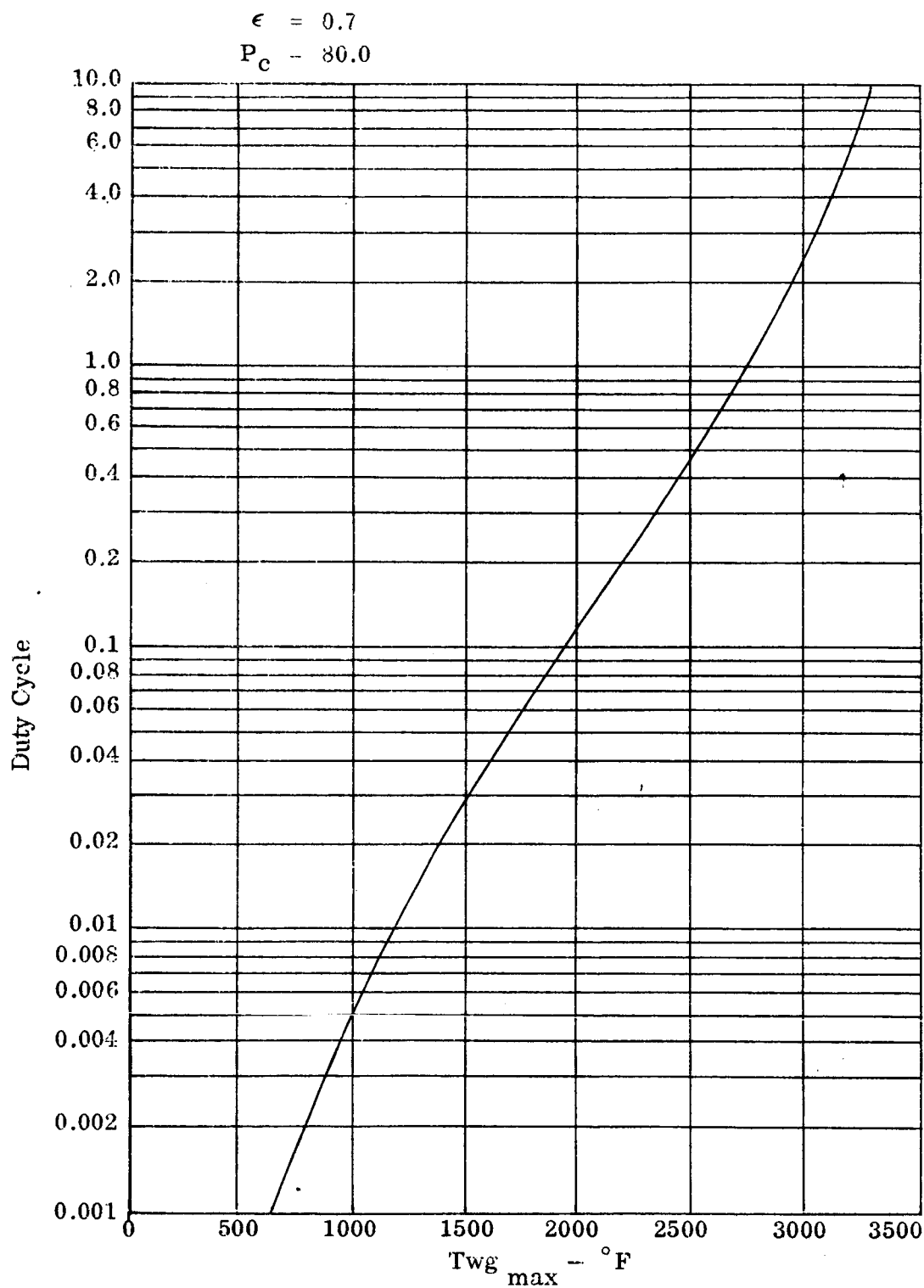


Figure V-24. Maximum Wall Temperature versus Duty Cycle

$$\bar{Q} = 0.7 Q \text{ Max}$$

1 hour ON Time

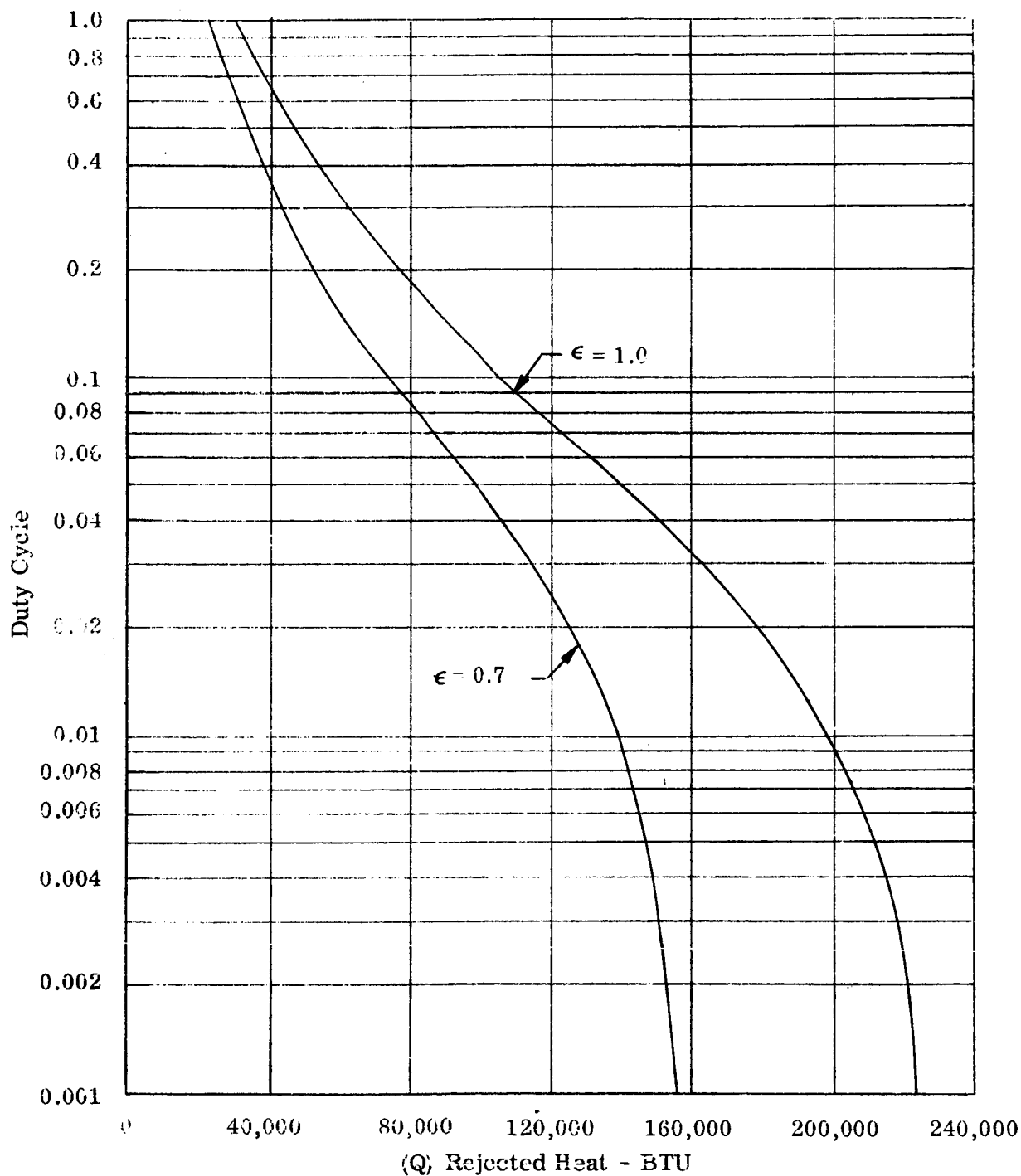
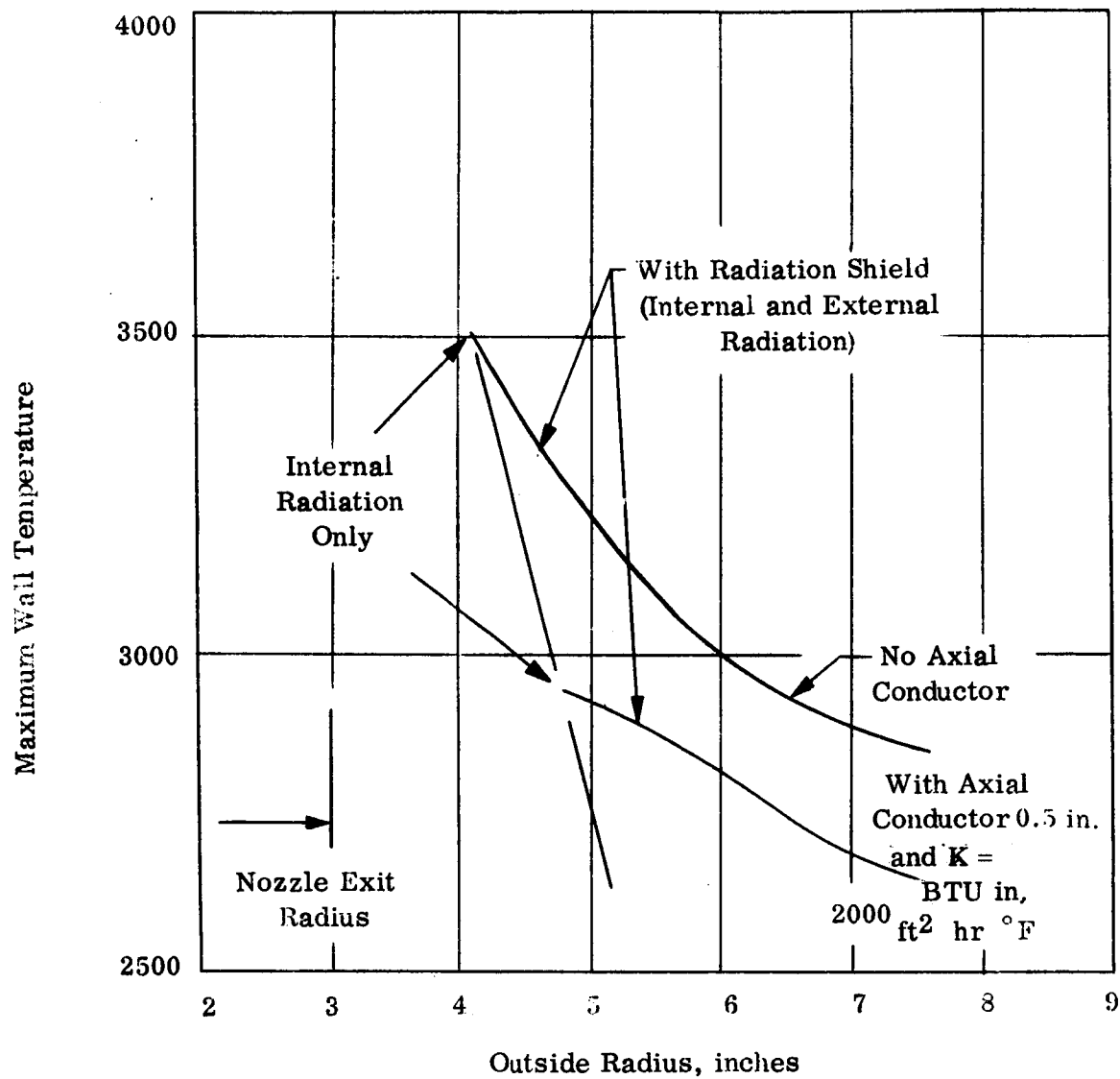


Figure V-25. Heat Rejected versus Duty Cycle



$T_G = 3500^\circ\text{F}$
 Radiation Shield
 Shape = Circular Cylinder
 $F = 100\text{ lb}$
 Chamber Pressure = 80 psia

Figure V-26. Maximum Wall Temperature versus Exit Radius - Radiation Cooled Concept

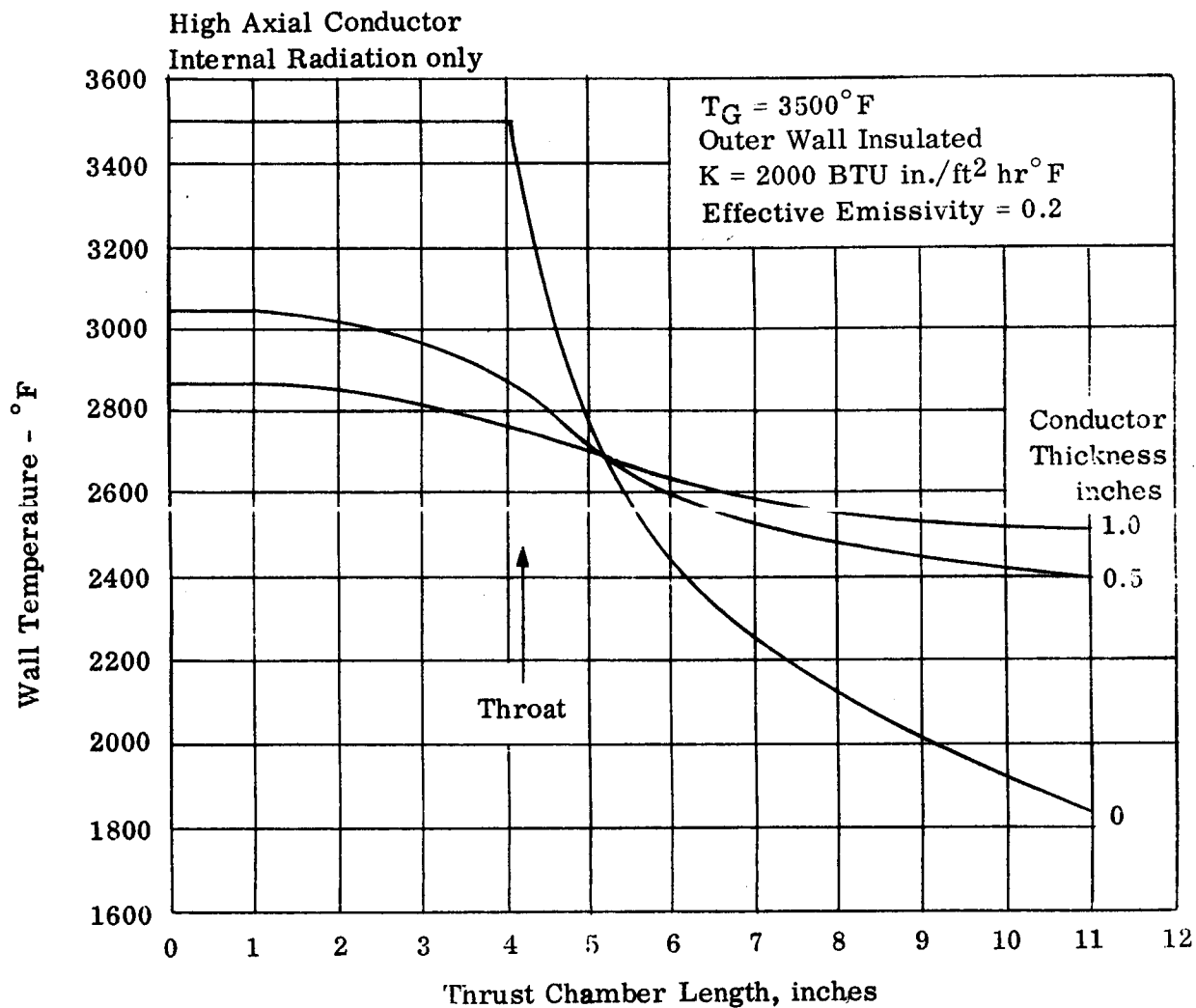


Figure V-27. Wall Temperature versus Axial Length - High Axial Conductor - 100 Pound Chamber

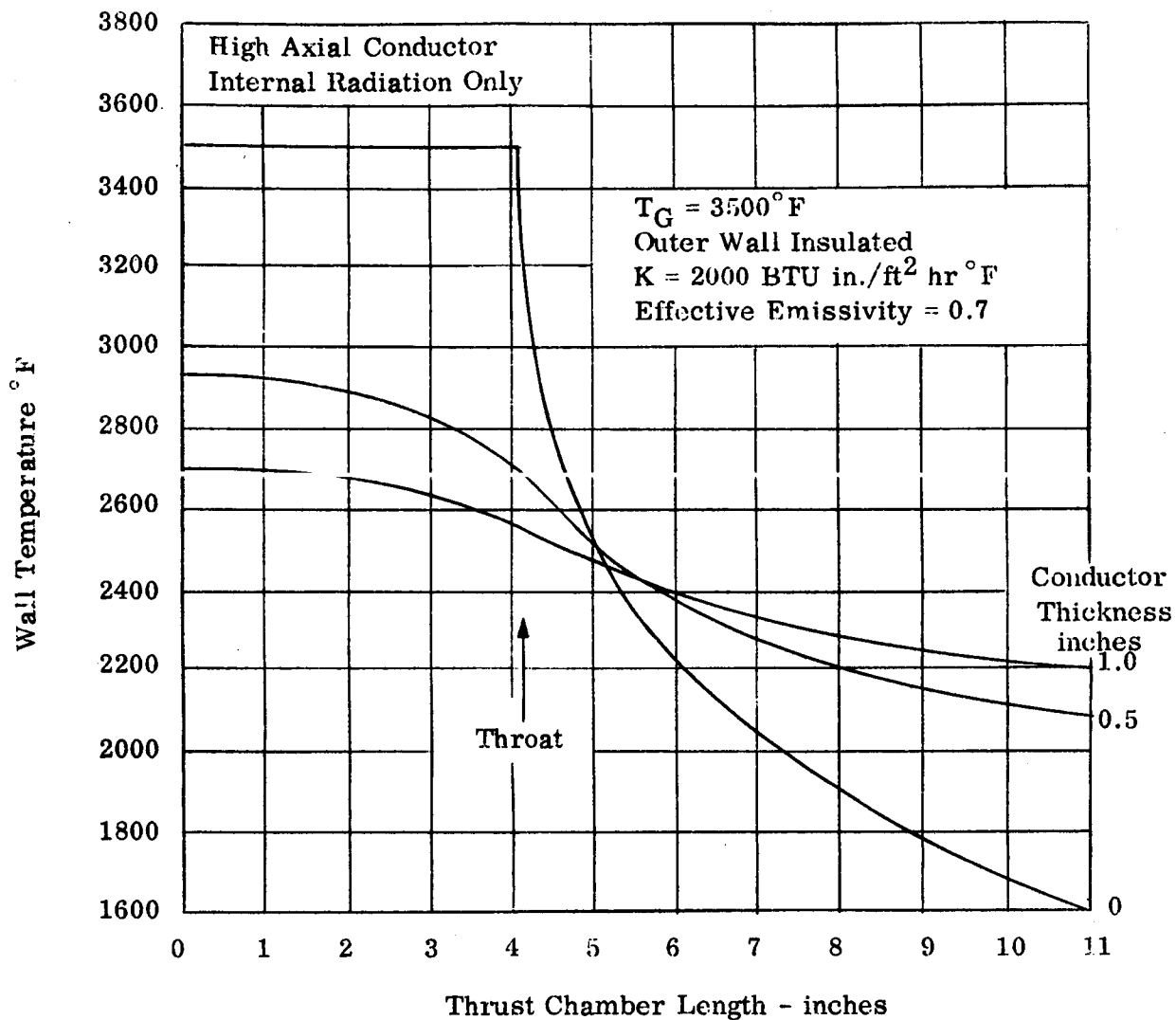


Figure V-28. Wall Temperature versus Axial Length - High Axial Conductor - 100 Pound Chamber

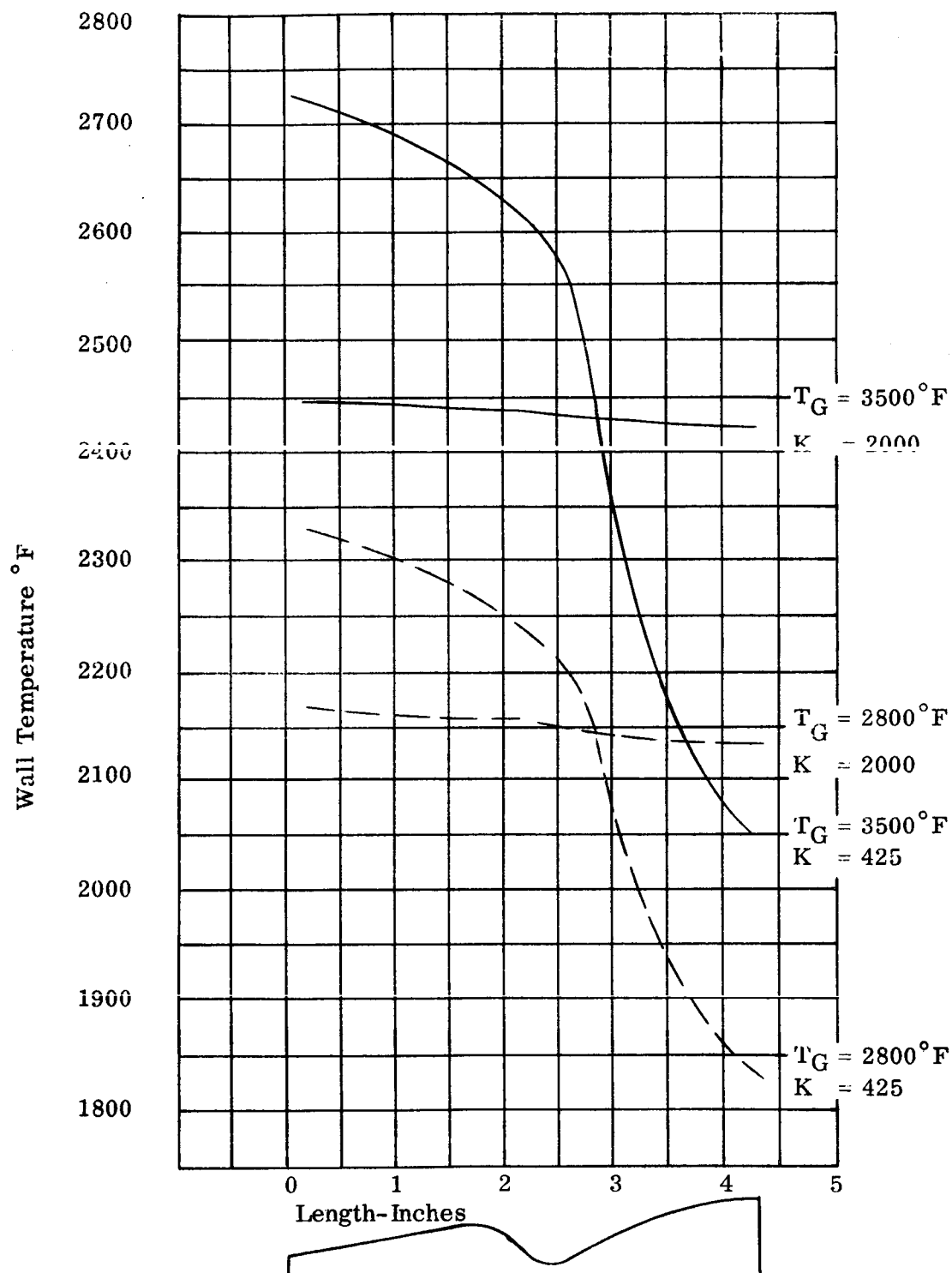


Figure V-29. Wall Temperature versus Axial Length - High Axial Conductor-
5 Pound Thrust Chamber Assembly

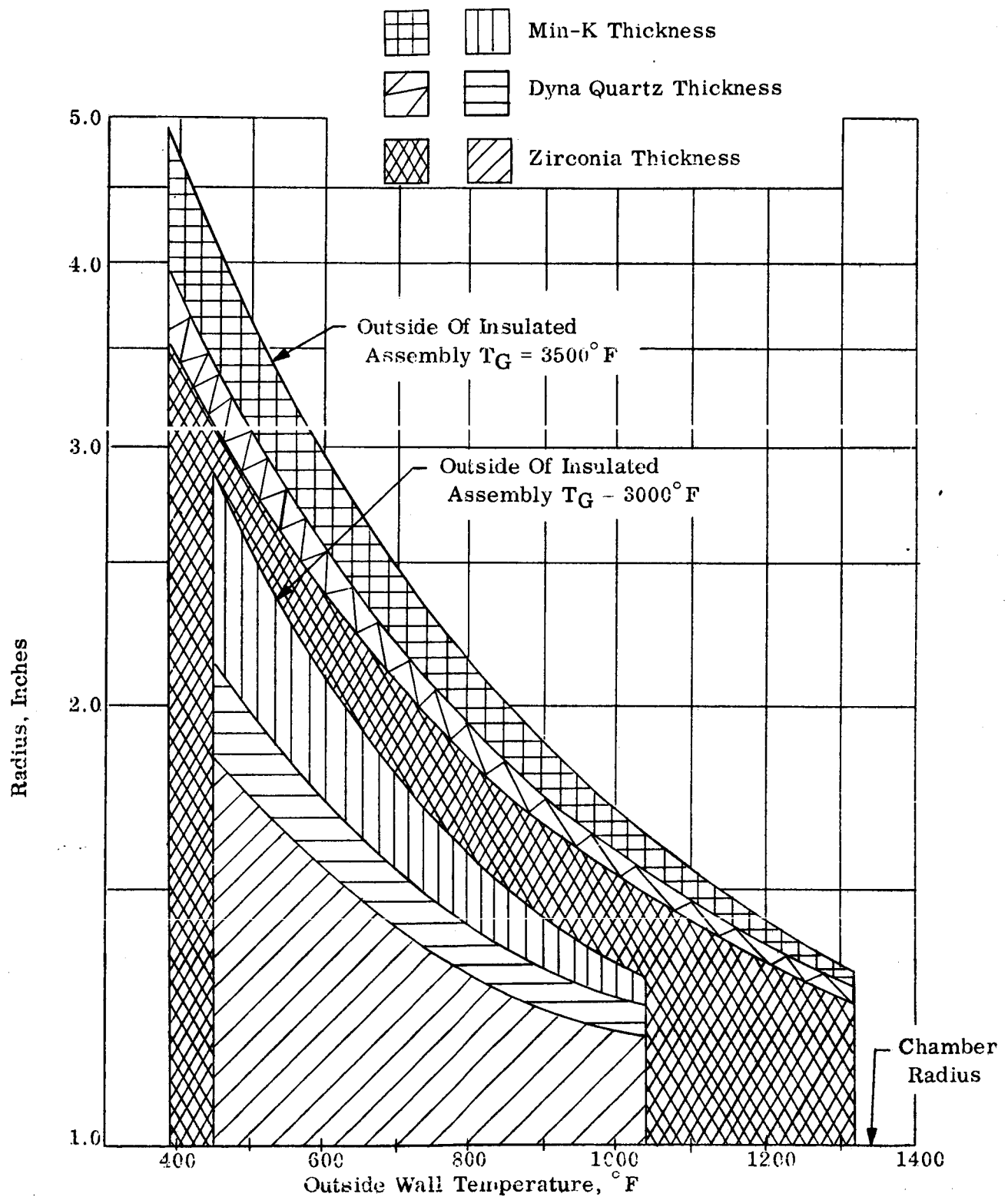


Figure V-30. Radial Thickness versus Outside Temperature - Composite Insulator

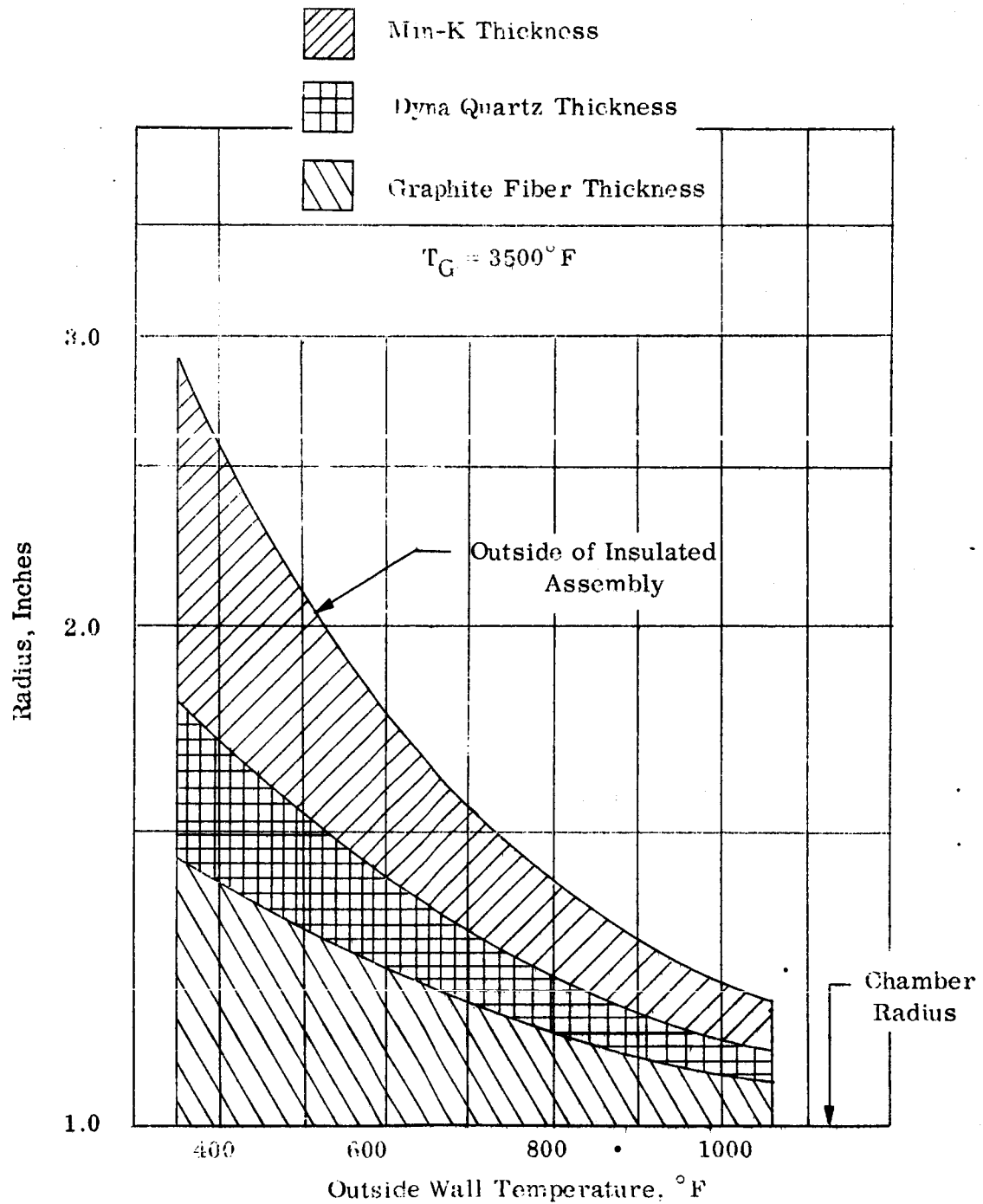


Figure V-31. Radial Thickness versus Outside Temperature - Composite Insulator

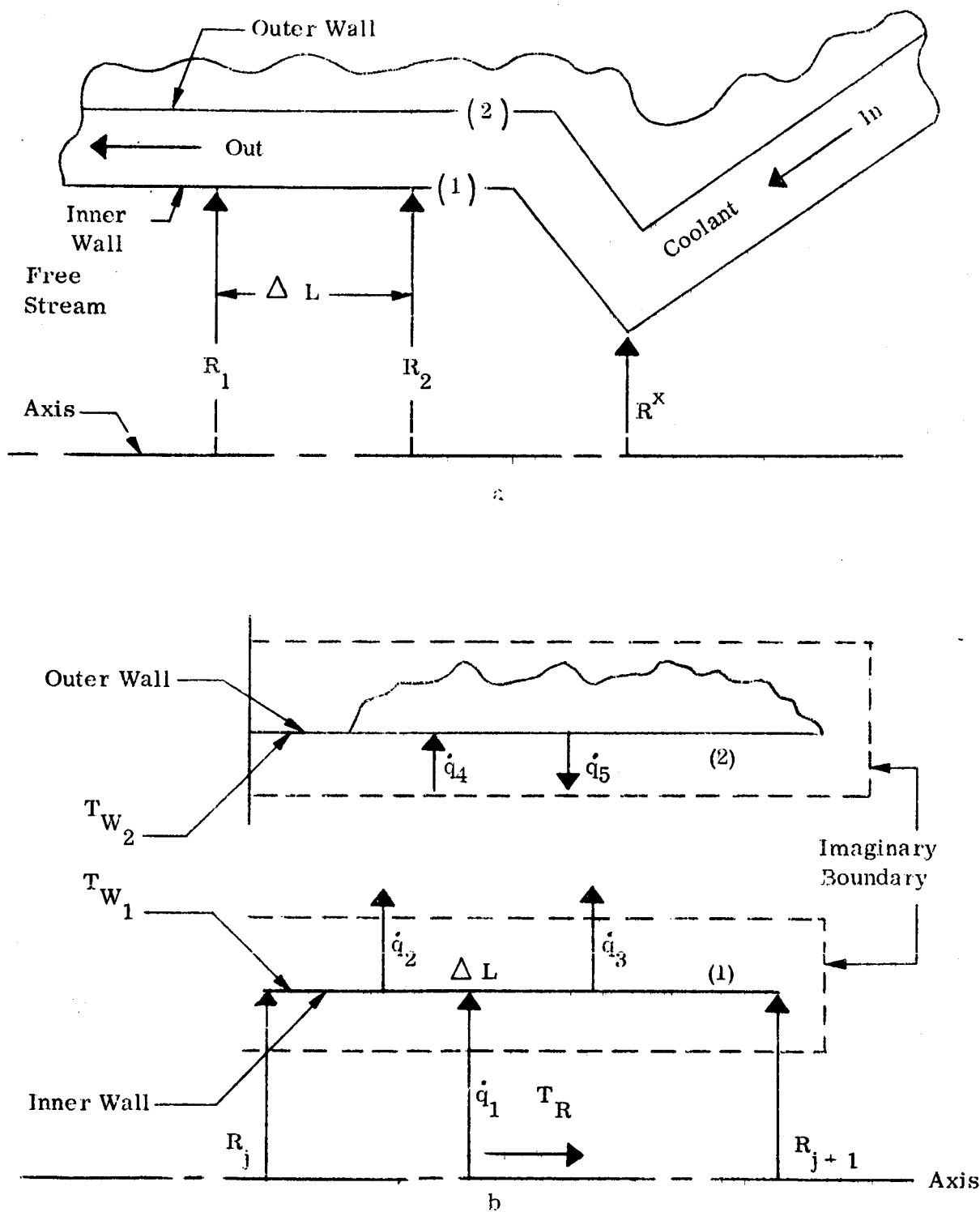


Figure V-32. Gas Regenerative Cooling

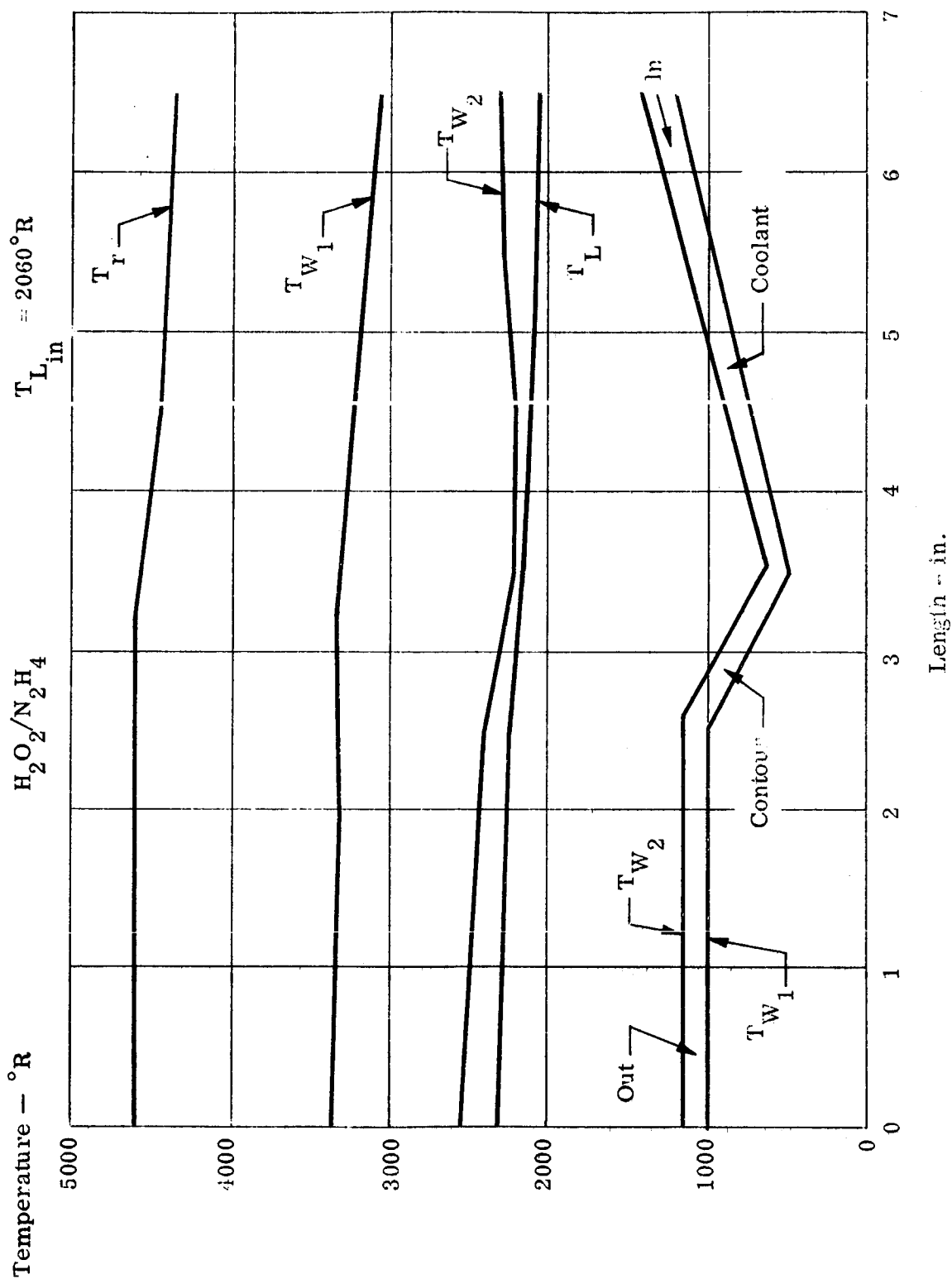


Figure V-33. Temperature versus Chamber Length, Gas Regenerative

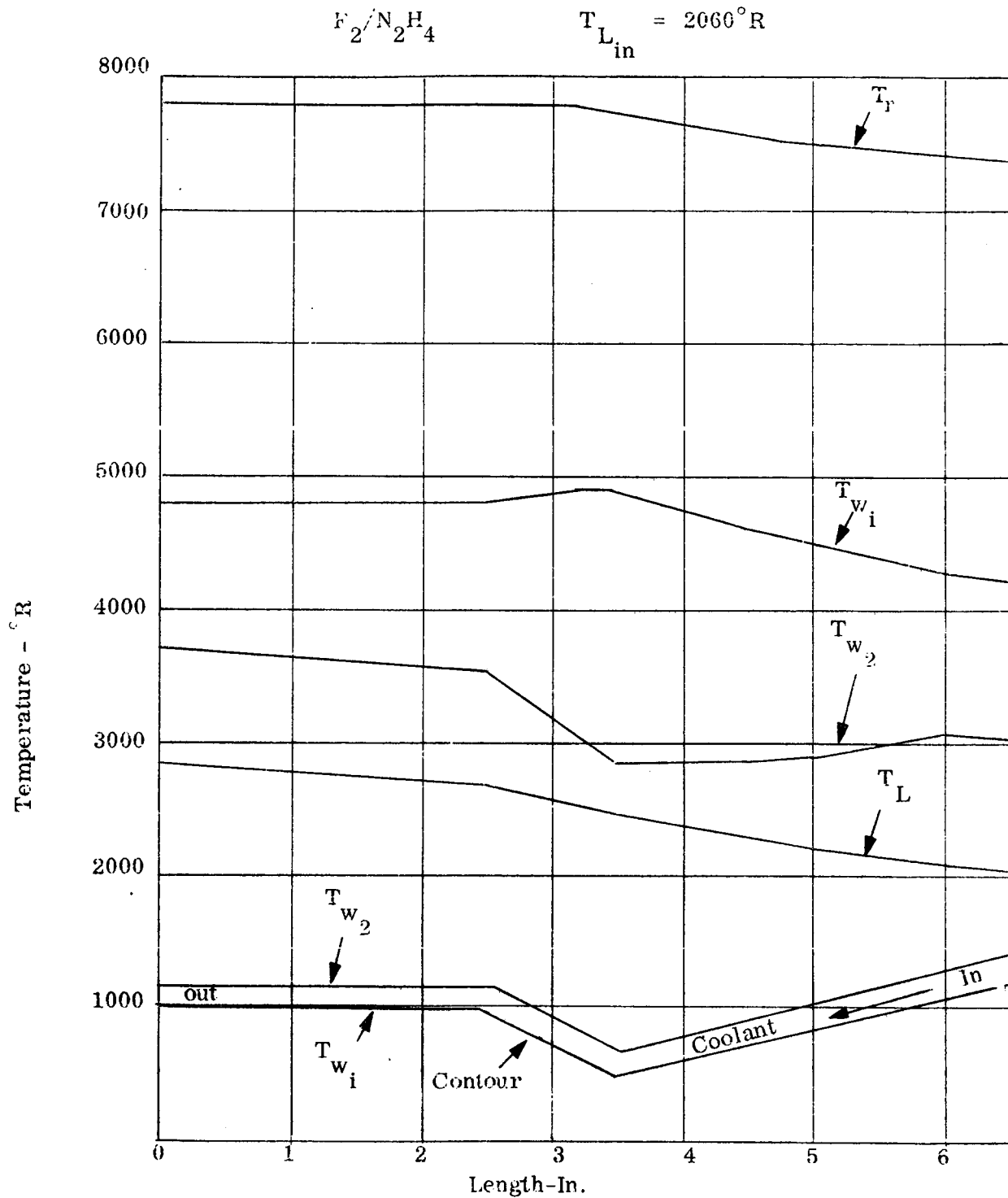


Figure V-34. Temperature versus Chamber Length, Gas Regenerative

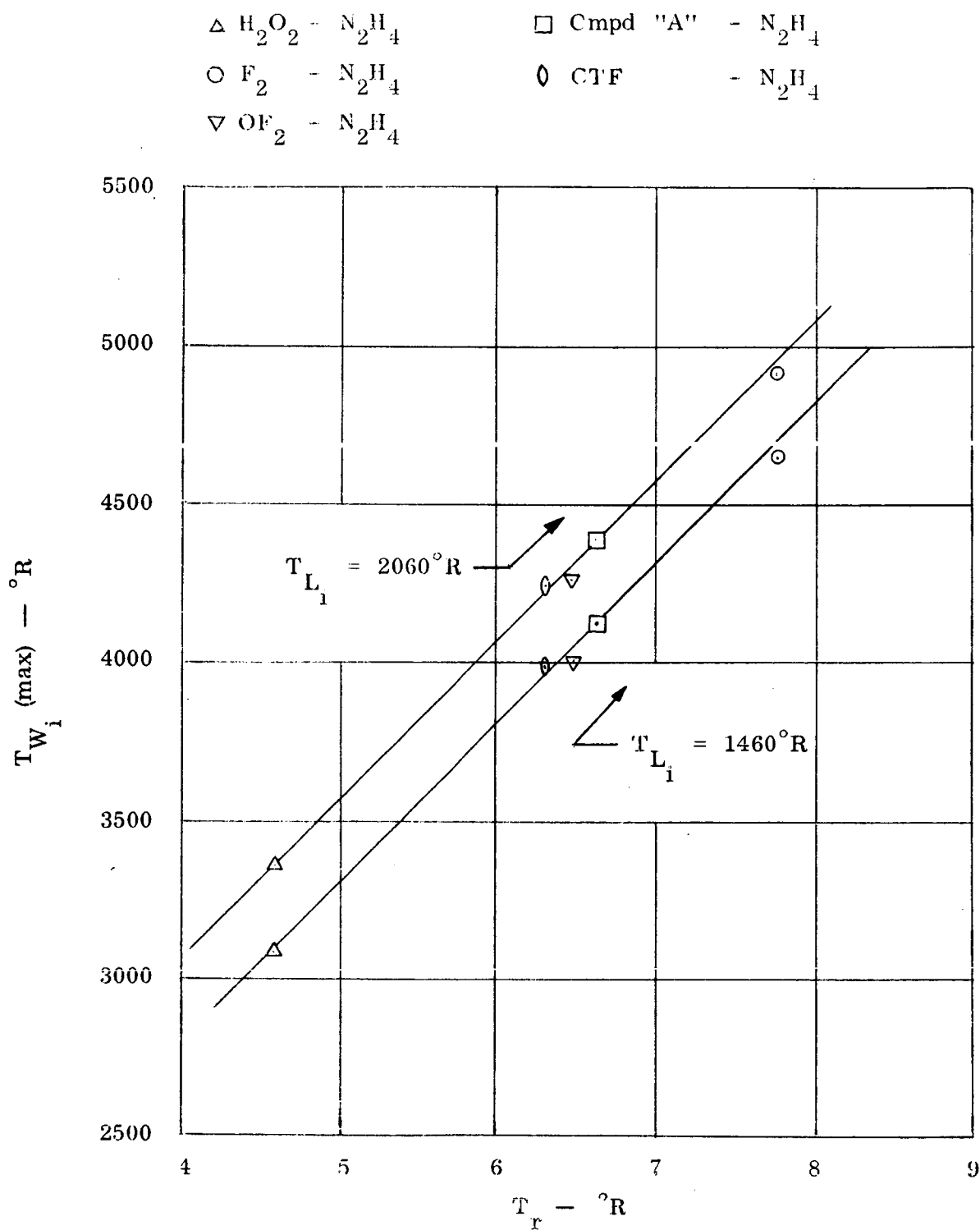


Figure V-35. Wall Temperature (max) versus Driving Gas Temperature Gas Regenerative

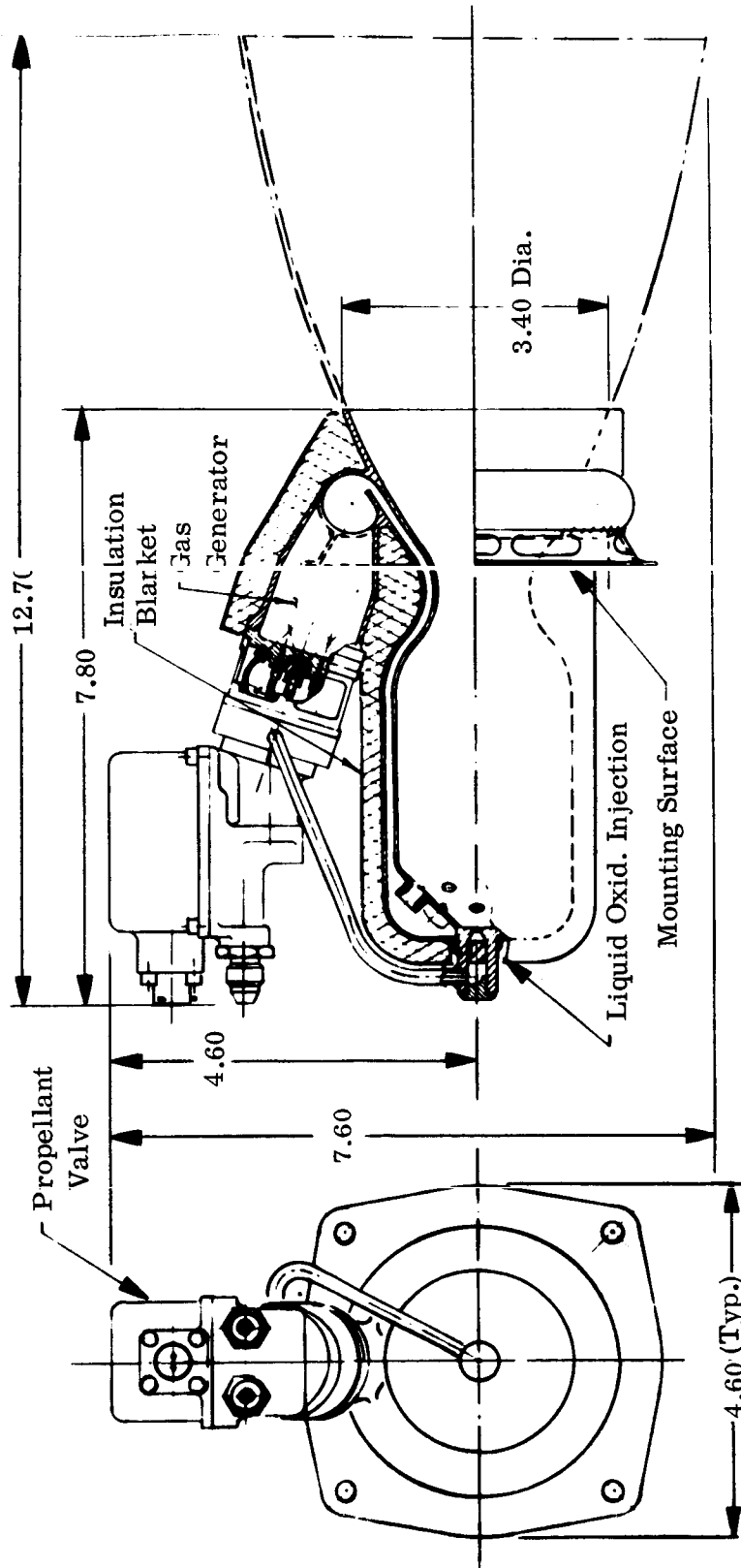


Figure V-36. Chamber Design Concept - Gas Regeneratively Cooled

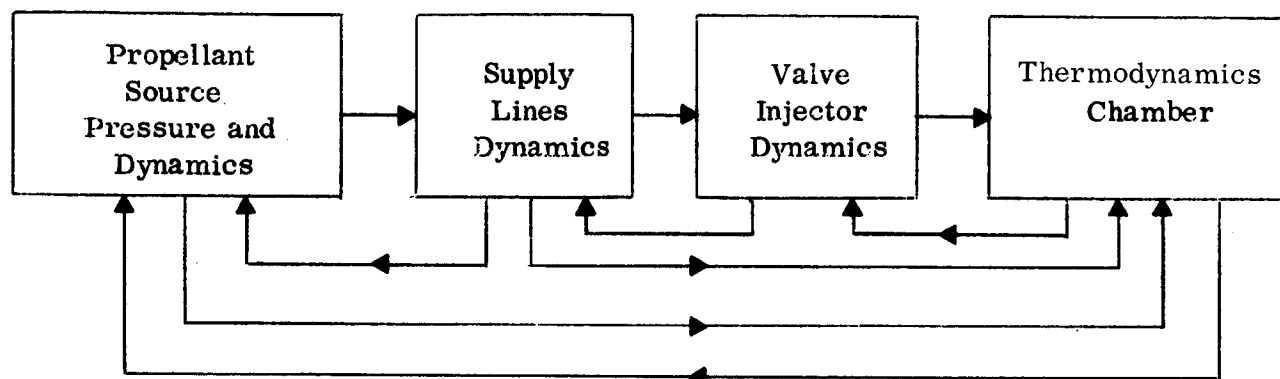
C. DYNAMIC BEHAVIOR ANALYSIS

An analytical model was arrayed to investigate system dynamic behavior and the necessary equations suitable for analog simulations were established.

1. Introduction

Combustion chamber dynamics is coupled with the injector, control valve, and supply line dynamics. The associated time delays and time lags are also contributors, along with combustion thermodynamics and kinematic properties. To incorporate all these contributing sources represents a formidable task. In the early part of this study efforts were expended toward obtaining suitable mathematical models which would provide an adequate description of the physical system. A vast amount of work has been done in this field by several investigators over the past several years. In all instances, it became necessary for each author to make certain assumptions which effectively simplified the mathematical equations to the extent that approximate solutions could be obtained. In the present study, it was found necessary to make certain assumptions which would be compatible with program time limitations. However, in setting up the mathematical model, care was exercised to orient the equations and computer diagrams in a manner which would exhibit growth potential. This growth potential produces the capability of investigation of new theories which may be substituted in block form into the present basic computer circuit. This block concept is necessary since it is imperative that in all cases the pressure drops must be compatible. This compatibility is assured by satisfying the appropriate pressure drop equations.

A simplified block diagram is given below.



The above diagram implies that it is impossible to disturb one element or block without affecting one or all others.

2. General Discussion

A brief discussion of each block shown in the diagram will now be given, with the intention of providing an indication of the present and projected scope of this program.

a. Propellant Source

The dynamic effects of the propellant source on the overall system response will be discussed. In the present system, the propellant is assumed to be under pressure. There is a pressure regulator in the system whose function is to maintain a constant supply pressure at the exit of the propellant source. The propellant source is subjected to reflected pressure waves generated by the opening and closing of the supply valve. To maintain constant pressure the frequency band of the pressure regulator must be of sufficient width to adequately accommodate the applied valve command frequency in the pulse mode. It becomes apparent at this point that the possibility of liquid-gas resonance in the propellant tank is possible.

Equations governing this phenomenon may be derived in a general manner and coupled with other portions of the system. These equations are assumed to be of secondary importance in this analysis and will not be pursued further at the present time. Incorporating this affect at a later date will not be particularly difficult

and may be encompassed in the growth potential of the present program. For simplicity alone it is assumed that the supply pressure is constant at all frequencies.

b. Supply Line Dynamics

Supply line dynamics may be adequately evaluated by use of "water hammer" theory. This theory has been well developed and may be employed to determine the dynamics of the fluid at the input of the control valve. This input source variation is important since a frequency does exist at which valve "starvation" will occur. This starvation phenomenon will exhibit itself when the supply line reflected pressure wave is 180° out of phase with the command valve input frequency conversely, when the pressure wave is in phase, then a resonance occurs in which the apparent valve inlet pressure will be greater than that assumed in the normal steady state condition. It now becomes apparent that a line transfer function is necessary to describe the dynamic effects on pressure drop and subsequent flow rates.

(1) Derivation of Supply Line Transfer Function

The fluid is assumed to be flowing in a pipe of uniform cross section, and the flow is laminar. The momentum and continuity equations governing flow of a fluid are derived in Reference (1) and are given as:

$$\frac{\partial u}{\partial x} = -\alpha \frac{\partial p}{\partial t} \quad (1)$$

$$\frac{u}{t} = -g \frac{\partial p}{\partial x} \quad (2)$$

where

$$\alpha = \rho g \left[\frac{1}{K} + \frac{2b}{fE} \right]$$

The sonic speed in the pipe is given by

$$a = \sqrt{\frac{g}{\alpha}} \quad (3)$$

Equations (1) and (2) will now be solved by use of LaPlace transforms. Let $U(x, s)$ and $P(x, s)$ be LaPlace transforms of $u(x, t)$ and $p(x, t)$,

respectively, and s is the LaPlace transform variable. The pipe is shown schematically in Figure V-37 as follows:

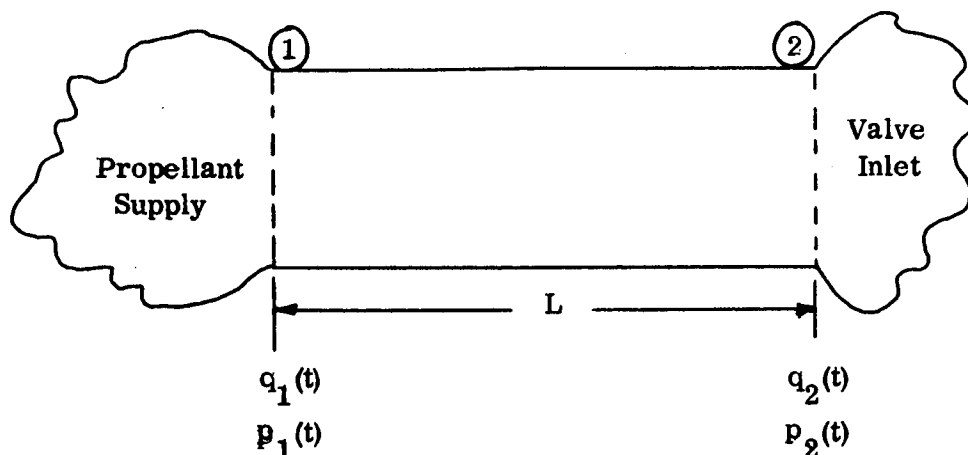


Figure V-37. Pipe Schematic

The average flow deviation at any cross section is given as

$$q(x, t) = A u(x, t) \quad (4)$$

The transforms of the time functions shown in Figure V-37 may be written as:

$$\begin{aligned} (a) \quad q_1(s) &= Q(0, s) \\ (b) \quad q_2(s) &= Q(L, s) \\ (c) \quad p_1(t) &= P(0, s) \\ (d) \quad p_2(t) &= P(L, s) \end{aligned} \quad (5)$$

A line impedance Z_0 and time constant β are defined as,

$$\begin{aligned} (a) \quad Z_0 &= \frac{a}{Ag} \\ (b) \quad \beta &= \frac{L}{a} \end{aligned} \quad (6)$$

The solutions of equations(1)and(2)subject to initial conditions and definitions given in equations(3),(5)and(6)are found to be,

$$Q_2(s) = Q_1(s) \cosh \beta s - \frac{P_1(s)}{Z_0} \sinh \beta s \quad (7)$$

$$P_2(s) = P_1(s) \cosh \beta s - Z_0 Q_1(s) \sinh \beta s \quad (8)$$

Equations (7) and (8) may be expressed in matrix form as

$$G(s) F_1 = F_2 \quad (9)$$

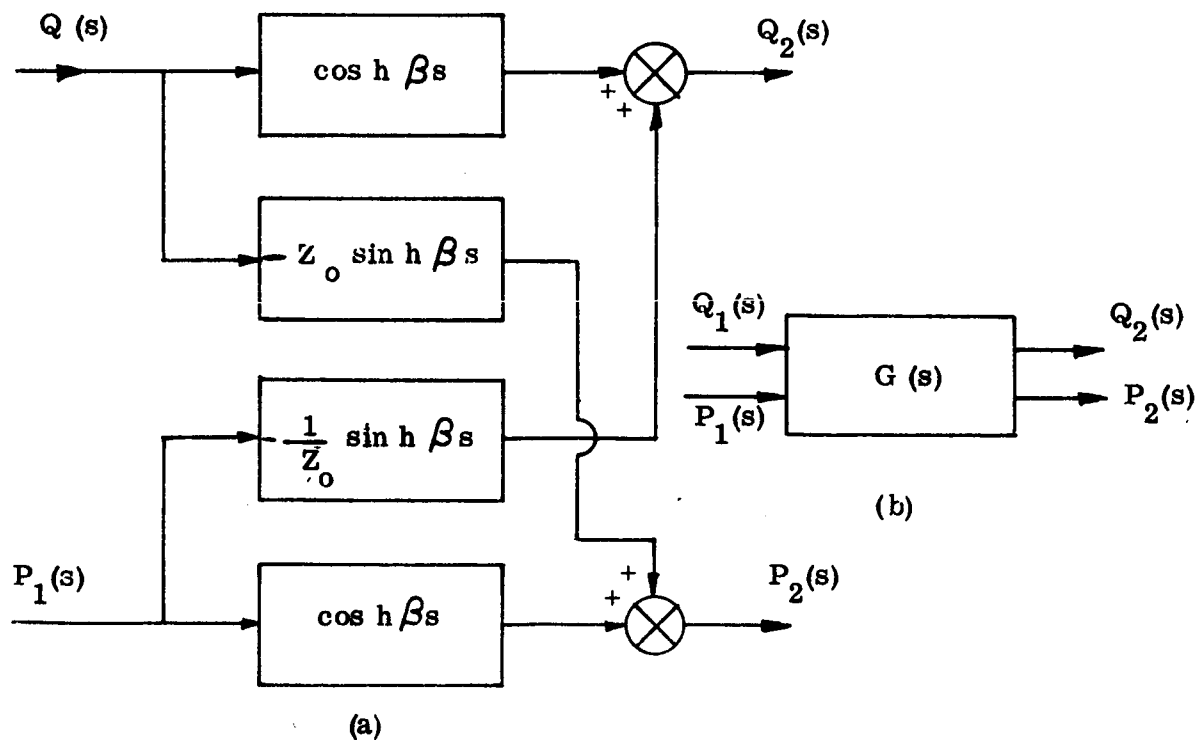
where

$$G(s) = \begin{vmatrix} \cosh \beta s & -\frac{1}{Z_0} \sinh \beta s \\ -Z_0 \sinh \beta s & \cosh \beta s \end{vmatrix}$$

$$F_1 = \begin{bmatrix} Q_1(s) \\ P_1(s) \end{bmatrix}$$

$$F_2 = \begin{bmatrix} Q_2(s) \\ P_2(s) \end{bmatrix}$$

A corresponding schematic block diagram is shown in the following block diagrams,



To proceed further with the mathematical development it is assumed that the propellant supply pressure at point① of Figure V-37, which represents entrance to the supply line, is held constant by a pressure regulator whose frequency response is broad enough to respond to all imposed frequencies. Point② of Figure V-37 is assumed to be the valve inlet and the exit area at this point is assumed to be proportional to a spool valve displacement $y(t)$. Since the pressure deviation at section① is zero,

$$p(0, t) = 0 \quad (10)$$

thus,

$$P_1(s) = 0$$

and equations (7) and (8) become,

$$Q_2(s) = Q_1(s) \cosh \beta s \quad (11)$$

$$P_2(s) = -Z_o Q_1(s) \sinh \beta s \quad (12)$$

Assume now that the flow through the valve is given as,

$$q(L, t) = C_1 y(t) + C_2 p(L, t) \quad (13)$$

The corresponding LaPlace transform of equation(13) becomes,

$$Q_2(s) = C_1 Y(s) + C_2 P_2(s) \quad (14)$$

where C_1 and C_2 are arbitrary constants.

Using equations (14) in (11) and (12) yields,

$$\frac{Q_1(s)}{Y(s)} = \frac{C_1}{\cosh \beta s + C_2 Z_o \sinh \beta s} \quad (15)$$

and,

$$\frac{P_2(s)}{Y(s)} = -Z_o C_1 \frac{\sinh \beta s}{\cosh \beta s + C_2 Z_o \sinh \beta s} \quad (16)$$

Equation (15) relates the flow rates of section① to valve position, and equation (16) relates the pressure at valve inlet to valve position.

The transfer functions given by equations (15) and (16) are proportional to quotients whose denominators are of the form

$$F(z) = \cosh z + B \sinh z = 0 \quad (17)$$

and whose roots represent the poles of the reference transfer functions.

To extract these roots define x and y to be the real and imaginary parts of z . Since

$$\cosh z = \frac{e^z + e^{-z}}{2}$$

$$\sinh z = \frac{e^z - e^{-z}}{2}$$

Equation (17) may be written as

$$e^{2x} \cos 2y + i e^{-2x} \sin 2y = \frac{B-1}{B+1} \quad (18)$$

Assuming B is real, equation (18) yields two equations.

$$e^{2x} \cos 2y = \frac{B-1}{B+1} \quad (19)$$

$$e^{2x} \sin 2y = 0 \quad (20)$$

The values of x and y satisfying equations (19) and (20) (with $n = 0, 1, 2 \dots$) are given by,

$$x = \frac{1}{2} \ln \left| \frac{B-1}{B+1} \right| \quad (21)$$

$$y = \pm \frac{2n+1}{2} \pi ; B < 1 \quad (22)$$

$$y = \pm n\pi ; B > 1 \quad (23)$$

The roots of equation (17) become,

$$z = \frac{1}{2} \ln \left| \frac{B-1}{B+1} \right| \pm i \frac{2n+1}{2} \pi ; B < 1 \quad (24)$$

$$z = \frac{1}{2} \ln \left| \frac{B-1}{B+1} \right| \pm i n\pi ; B > 1 \quad (25)$$

For $B = 1$ there are no bounded roots of equation (17).

Since the roots are known, equation (17) may be expressed as an infinite product. Assuming,

$$X_B = \frac{1}{2} \ln \left| \frac{B-1}{B+1} \right|,$$

equation (17) may be written as

$$\cosh z + B \sinh z = \prod_{n=0}^{\infty} \left[1 - \frac{z}{X_B + i \left(\frac{2n+1}{2} \pi \right)} \right] \left[1 - \frac{z}{X_B - i \left(\frac{2n+1}{2} \pi \right)} \right]; B < 1 \quad (26)$$

$$\cosh z + B \sinh z = \prod_{n=0}^{\infty} \left[1 - \frac{z}{X_B} \right] \prod_{n=0}^{\infty} \left[1 - \frac{z}{X_B + i n \pi} \right] \left[1 - \frac{z}{X_B - i n \pi} \right]; B > 1 \quad (27)$$

Multiplying the complex conjugate infinite terms in the infinite products yields,

$$\cosh z + B \sinh z = \prod_{n=0}^{\infty} \left[1 - \frac{2X_B z - z^2}{X_B^2 + \left(\frac{2n+1}{2} \pi \right)^2} \right]; B < 1 \quad (28)$$

$$\cosh z + B \sinh z = \left[1 - \frac{z}{X_B} \right] \prod_{n=0}^{\infty} \left[1 - \frac{2X_B z - z^2}{X_B^2 + n^2 \pi^2} \right]; B > 1 \quad (29)$$

Since the term $\sinh \beta s$ is required in equation (16) it must now be evaluated. Define $\sinh \beta s/s$ at $s = 0$ by,

$$\frac{\sinh \beta s}{\beta s} \bigg|_{s=0} = 1 \quad (30)$$

The equation,

$$\frac{\sinh \beta s}{s} = 0$$

has roots at,

$$\beta s = \pm i n \pi; n \neq 0,$$

which can also be derived from equation (24) by letting $B \rightarrow \infty$. It follows that,

$$\sinh \beta s = \beta s \prod_{n=1}^{\infty} \left[1 + \frac{\beta^2 s^2}{n^2 \pi^2} \right] \quad (31)$$

The supply line transfer function equations (15) and (16) may now be written as,

$$\frac{Q_1(s)}{Y(s)} = \prod_{n=0}^{\infty} \frac{C_1 \left[X_B^2 + \left(\frac{2n+1}{2} \right)^2 \pi^2 \right]}{\left[X_B^2 + \left(\frac{2n+1}{2} \right)^2 \pi^2 - 2 X_B z - z^2 \right]} \Bigg|_{z = \beta s} \quad B < 1 \quad (32)$$

$$\frac{P_2(s)}{Y(s)} = - \prod_{n=0}^{\infty} \frac{Z_0 C_1 z \left[1 + z^2/n^2 \pi^2 \right] \left[X_B^2 + \left(\frac{2n+1}{2} \right)^2 \right]}{X_B^2 + \left(\frac{2n+1}{2} \right)^2 \pi^2 - 2 X_B z - z^2} \Bigg|_{z = \beta s} \quad B < 1 \quad (33)$$

or

$$\frac{Q_1(s)}{Y(s)} = \prod_{n=0}^{\infty} \frac{C_1 \left[X_B^2 + n^2 \pi^2 \right]}{\left[1 - \frac{z}{X_B} \right] \left[X_B^2 + n^2 \pi^2 - 2 X_B z - z^2 \right]} \Bigg|_{z = \beta s} \quad B > 1 \quad (34)$$

$$\frac{P_2(s)}{Y(s)} = \prod_{n=0}^{\infty} \frac{Z_0 C_1 z \left[1 + z^2/n^2 \pi^2 \right] \left[X_B^2 + n^2 \pi^2 \right]}{\left[1 - \frac{z}{X_B} \right] \left[X_B^2 + n^2 \pi^2 - 2 X_B z - z^2 \right]} \Bigg|_{z = \beta s} \quad B > 1 \quad (35)$$

Equations (32) - (35) are the transfer functions of the supply lines. The line dimensions determines which set of equations are applicable by dictating whether $B \lesseqgtr 1$. The number of terms which must be retained is a function of accuracy requirements and the bandwidth of the applied frequency spectrum. The equations are amenable to analogue simulation since any number of terms may be "patched" in for improved accuracy.

c. Valve Dynamics

The equations used in describing the servo valve dynamics are those generally used in a pressure control system. The associated non-linear gain properties are preserved to adequately describe pressure drop due to propellant flow. The transfer function for a servo valve subjected to both blocked load and flow drop is given as

$$P(s) = \frac{K_1 i(s) - K_2 (1 + \tau s) Q(s)}{1 + \frac{2\zeta}{w_n} s + (s/w_n)^2} \quad (36)$$

Equation (36) represents a general transfer function for a valve with throttling capability. Since the experimental engine uses an on-off valve we assume $K_1 = \tau = 0$ to obtain

$$P(s) = \frac{K_2' Q(s)}{1 + \frac{2\zeta}{w_n} s + (s/w_n)^2} \quad (37)$$

d. Injector Dynamics

Injector dynamics is not currently amenable to mathematical treatment and as a result injector design must be based on experimental results. For lack of anything better perfect injection is assumed.

This assumption implies perfect atomization of the on-fuel combination for subsequent burning. Imperfect injection will be investigated when a proper mathematical description or model is available. Injector pressure drop will, however, be incorporated.

e. Time Delay Dynamics

A finite time is required for the fuel and oxidizer to travel from the valve to the combustion chamber. In general, however, the line impedance for each will not be the same. Hence, one will arrive before the other resulting in an off mixture ratio at the beginning of each pulse. This effect will be investigated by estimating the transport time difference and incorporating it in the simulation. This is being accomplished by inserting a relay in the simulated oxidizer and fuel flow lines set to close at prescribed time intervals after valve command signal is given.

f. Combustion Time Lag

The combustion time lag is characterized by a first order time lag function. The function is assumed to delay the chamber pressure rise by a time constant τ_c , or

$$p_c = \frac{p_{c1}}{\prod_c s+1} \quad (38)$$

g. Thermodynamics

The thermodynamic process is assumed to be governed by the perfect gas laws. A different model is under investigation and will be ready for analysis at a later date. The gas law may be written as,

$$p = \frac{1}{v} m R T \quad (39)$$

Assuming that p is perturbed by an amount Δp we have (assuming volume is constant)

$$\Delta p = \frac{\partial p}{\partial m} \Delta m + \frac{\partial p}{\partial r} \Delta R + \frac{\partial p}{\partial t} \Delta T \quad (40)$$

Using equation (39) yields

$$\frac{\partial p}{\partial m} = \frac{RT}{v}; \quad \frac{\partial p}{\partial R} = \frac{mT}{v}; \quad \frac{\partial p}{\partial T} = \frac{mR}{v}$$

and equation (40) may be written as

$$\frac{\Delta p}{\Delta t} = \dot{p}_c = \frac{RT_c}{v} \dot{m} + \frac{mT_c}{v} \dot{R} + \frac{mR}{v} \dot{T}_c \quad (41)$$

But

$$\dot{T} = \frac{dT}{dt} = \frac{\partial T_c}{\partial r} \dot{r} + \frac{\partial T}{\partial p} \dot{p}_c \quad (42)$$

also

$$r = \frac{\dot{m}_{oT}}{\dot{m}_{fT}} \quad (43)$$

and since total flow equals flow in chamber plus flow in nozzle,

$$\dot{m}_T = \dot{m}_c + \dot{m}_e$$

or

$$\dot{m}_T = \dot{m}_{fT} (r+1) \quad (44)$$

similarly

$$\dot{m}_c = \dot{m}_{fc} (r+1) \quad (45)$$

$$\dot{m}_c = \frac{A_t p_c}{c^*} g \quad (46)$$

Using equations (42) and (45) in equation (41)

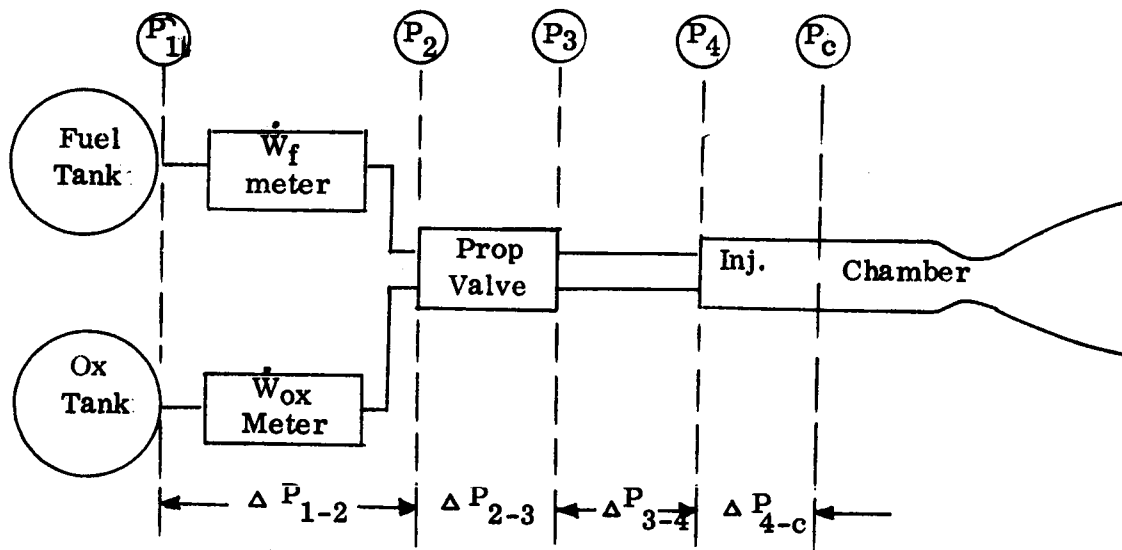
$$\left(1 - m_{f_c} \frac{R(r+1)}{v} \frac{\partial T}{\partial p_c}\right) \dot{p}_c = \frac{R(r+1)}{v} \left[\dot{m}_{f_c} T + \dot{m}_{f_c} \frac{\partial T}{\partial r} \dot{r} \right] \quad (47)$$

Equation (47) gives chamber pressure as a function of the thermodynamic properties of the mixture.

The partial derivatives used in equation (46) were obtained from Reference (4).

h. Pressure Drop Equations

Definition of pressure drops are shown schematically below:



Using the above sketch and assuming pressure drops are due to flow rates and accelerated flow, the total pressure drop is given as,

$$p_1 - p_c = (p_1 - p_2) + (p_2 - p_3) + (p_3 - p_4) + (p_4 - p_c) \quad (48)$$

where

$$p_1 - p_2 = K_1 \dot{w}^2 + K_2 \ddot{w} \quad (49)$$

$$p_2 - p_3 = \frac{K_2 Q}{(s/w_n)^2 + 2\zeta s/w_n + 1} = \text{valve pressure drop} \quad (50)$$

$$p_3 - p_4 = K_5 \dot{w}^2 + K_6 \ddot{w} \quad (51)$$

$$p_4 - p_c = K_7 \dot{w}^2 + K_8 \ddot{w} \quad (52)$$

Using equations (49), (50), (51), and (52) in equation (48), and solving for w ,

$$\ddot{w} = \frac{1}{K_2 + K_6 + K_8} \left[(p_1 - p_c) - (K_1 + K_5 + K_7) \dot{w}^2 - \frac{K_2 Q}{(s/w_n)^2 + 2\zeta(s/w_n) + 1} \right] \quad (53)$$

Thus $\dot{w}$ or $\dot{m}$ may be obtained by integrating equation (52).

The line impedances and accelerated flow constants have been estimated for a given configuration. The valve pressure drop characterized by K_2 in equation (53) are determined experimentally and given as Figures V-38 and V-39 for oxidizer and fuel, respectively.

3. Integrated System

The system equations were derived in the previous section. For convenience, the system is assumed to be divided into four major divisions.

- (1) Fluid (water hammer) effects
- (2) Pressure drop - flow
- (3) Valve and combustion chamber dynamics
- (4) Thermodynamics

Each of these divisions will require modification to finally evolve a model which will exhibit certain dynamic properties observed during experiment.

Due to problem complexity, it was found convenient to assume that each pulse is independent of the other. This definitely is not true in the transient, but is true in the steady state provided the pulse repetition rate is not too high and fluid (water hammer) effects are neglected. Since at the present time a single pulse will be simulated, the fluid (water hammer) effect has not been simulated.

Figure V-40 shows block diagrams of fuel valve and line time delay dynamics (the oxidizer valve is identical). Here, the valve is not the on-off type actually simulated, but rather the general purpose valve with throttling capability.

Figure V-41 shows a block diagram of the thermodynamics loop.

4. Summary of Equations

The equations which have been simulated for analogue study are given as follows:

(a) Pressure drop-flow

$$\ddot{w} = \frac{1}{K_2 + K_6 + K_8} \left[p_1 - p_c - (K_1 + K_5 + K_6) \dot{w}^2 - \frac{K_2 Q}{\left(\frac{s}{w_n}\right)^2 + 2\left(\frac{s}{w_n}\right) + 1} \right]$$

(b) Combustion chamber dynamics

$$p_c = \frac{p_{c1} e^{\alpha_s}}{\tau_c s + 1}$$

(c) Thermodynamics

$$(1) \quad (1 - m_{f_c} \frac{R(r+1)}{v} \frac{\partial T}{\partial p_c}) \dot{p}_c = \frac{R(r+1)}{v} \left[\dot{m}_{f_c} T + m_{f_c} \frac{\partial T}{\partial r} \dot{r} \right]$$

$$(2) \quad \dot{m}_T = \dot{m}_c + \dot{m}_e$$

$$(3) \quad \dot{m}_{f_c} = \dot{m}_{f_t} - \frac{A_t}{c^*(r+1)} p_c$$

$$(4) \quad \dot{m}_{f_c} = \int \left(\dot{m}_{f_t} - \frac{A_t}{c^*(r+1)} p_c \right) dt$$

$$(5) \quad \dot{m}_c = \frac{A_t}{c^*(r+1)} p_c$$

$$(6) \quad \frac{\partial T}{\partial p_c} = f(r)$$

$$(7) \quad T = \int \dot{p}_c f(r) dt + T_0$$

5. General Theoretical Approach

A general approach used in mathematical physics is based on the assumption that total change, or variation, of a function of several variables is proportional to the sum of partial changes of the variables. Applying this theory to the pressure induced in a combustion chamber yields:

$$dP_c = \frac{\partial P_c}{\partial M} dM + \frac{\partial P_c}{\partial R} dR + \frac{\partial P_c}{\partial T} dT + \frac{\partial P_c}{\partial x} dx + \frac{\partial P_c}{\partial r} dr + \frac{\partial P_c}{\partial o} do + \frac{\partial P_c}{\partial e} de \quad (54)$$

If we assume: (a) homogeneous gas distribution throughout chamber, (b) perfect gas laws are applicable, and (c) reflected pressure waves are negligible; then last four terms of equation (54) may be neglected and the equation agrees with equation presented in Reference

1. It was further shown that if $P_c = P_c(t)$ then the time rate of change $\dot{P}_c$ is given as:

$$\left[1 - M_{fc} \frac{R(r+1)}{V} \frac{\partial T}{\partial P_c} \right] \dot{P}_c = \frac{R(r+1)}{V} \left[\dot{M}_{fc} T + M_{fc} \frac{\partial T}{\partial r} \dot{r} \right] \quad (55)$$

$$\text{It was also assumed that, } \dot{M}_{fc} = \dot{M}_{ft} - \frac{A_t g}{c^*(r+1)} P_c \quad (56)$$

$$\text{using equation (3) in equation (2) yields } \tau_c \dot{P}_c + P_c = F \quad (57)$$

$$\text{where } \tau_c = \frac{1 - M_{fc} \frac{R(r+1)}{V} \frac{\partial T}{\partial P_c}}{\frac{RT A_t g}{V c^*}} \quad (58)$$

$$F = \frac{T \dot{M}_{ft} + M_{fc} \frac{\partial T}{\partial r} \dot{r}}{T \frac{A_t g}{c^*(r+1)}} \quad (59)$$

Note that equation (57) is a first order equation in P_c . If, during some process τ_c and F are constant, then the response $P_c(t)$ will be that of a first order time lag system.

The actual value of τc , consistent with the present model, is given by equation (58). Note that under certain conditions ($\tau c \leq 0$) with a negative time constant, $P_c(t) \rightarrow \infty$. This presents an interesting situation since $\tau c = f(M_{fc}, \frac{\partial T}{\partial P_c}, r)$ and since each of these variables are functions of time, it is possible for τc to change sign during a time variation of P_c . This clearly indicated that, even with this simplified model, it is possible to simulate a spike which has been observed experimentally. Some possible mechanisms for generation of large parameter values will now be presented.

- (a) The M_{fc} can be made large by a combustion time delay. This delay is physically realizable since a finite time is required for vaporization and subsequent burning of the propellants. Other reasons might possibly be given.
- (b) The r term (mixture ratio) can be made large by allowing excess oxidizer flow.
- (c) The $\frac{\partial T}{\partial P_c}$ term appears to be the adiabatic temperature rise due to change in chamber pressure. However, it might also be used to describe possible freezing of the propellants when entering a chamber with zero pressure.

For generation of a pressure spike a possible variation of τc is shown as Figure V-42. Figure V-42a reveals that for pressure spike generation τc must be initially negative or if zero it must go negative to some minimum value and decrease to zero in some characteristic time, t^* . Figure V-42b reveals that the magnitude of $P_{c_{max}}$ is functionally related to t^* and if the chamber is to survive the initial spike then t^* cannot be larger than some critical value $t^* = t_a^*$.

Figure V-42a further reveals that the oscillations observed experimentally on $P_c(t)$ may be generated by an oscillation of τc about its required value.

When a train of pulses is presented and the effect of initial chamber temperature and residual pressure are determined, it may possibly be shown that τc for pulse numbers larger than "n" will always be positive during the pulse.

Similar reasoning shows that the observed characteristics may be simulated with a second order equation where the frequency and damping are functions of the chamber and thermodynamic parameters.

6. Simulation Results

The model yields steady state results and is shown to be a good candidate for modification to incorporate effects whose presence attributes to instability or other adverse flow and pressure disturbances.

The partial derivative terms in the equations were computed from theoretical combustion data. These terms are important since their presence allow the use of perturbation theory over a large range of variables.

For analogue simulation it was felt that a 1000:1 time scale would be adequate. The equations were written in machine language and patch board wiring for the PACE computer was accomplished. The partial derivative terms were scaled properly and mechanized on the function generators in preparation for problem solutions.

Some typical results will now be presented and discussed. Figure V-43 shows a typical run. Here the line size has been adjusted to obtain rated flow for a supply tank pressure of 200 psi, and a desired chamber pressure of 80 psi. The time required to obtain steady state flow rate is a function of the acceleration constant of the propellant supply lines. Several runs were made for several acceleration constants to determine the nature of this effort. The results are shown in Figure V-44. The variation can be extended until an overshoot occurs in the P_c trace shown in Figure V-43. This figure further implies that the time required to obtain steady state P_c is linear in the range investigated. This appears to be true for both oxidizer and fuel lines.

The inherent flexibility of an analogue simulation system allows one to obtain at least qualitative effects upon system response by rather simple techniques. The effect of a time delay was simulated by use of comparators on the analogue computer. A circuit was mechanized which would allow the installation of a time delay in the oxi-

dizer circuit. The results of this phase of the investigation is interesting in that the P_c has a characteristic noted in experimental tests where P_c rises rapidly and apparently the chamber tends to starve causing a decrease in P_c , a secondary spike. A typical run is shown in Figure V-45. It was further noted that the time to steady state was affected by this time delay. Several runs were made varying the time delay. These results are shown in Figure V-46, indicating that for the given configuration a time delay less than 0.002 sec produces no secondary spike. For time delays greater than this value the magnitude of the spike is practically linear with the magnitude of the delay.

The underlying philosophy of these investigations is to determine what conditions are necessary to produce characteristics observed during tests. The present very simple model is capable of producing the two aforementioned characteristics. Expressing these variables in terms of physical properties remains a future task.

Another adverse characteristic observed during tests is the occurrence of an initial pressure spike. To produce this spike the condition of combustion time delay was simulated. This was accomplished by inserting a delay circuit in the P_c portion of the computer loop. A typical run is shown in Figure V-47. The ignition delay is 0.0002 sec and the P_c spike magnitude is 400 psi. Note that the P_c spike retards the oxidizer and fuel flow rates. It may be further noted that mixture ratio varies considerably during the transient interval of chamber pressure variation.

The discussion presented thus far assumes instantaneous combustion, and that the transportation time delay is zero. In attempting to solve linear differential equations incorporating a time delay, serious mathematical difficulties are encountered. This is true for both the time and frequency domains. Introducing the transport time delay implies that the effect of P_c at time, t , is a function of the magnitude of P_c at τ seconds earlier. This condition is assumed to be continuous over the interval P_c . Equation (4) modified to incorporate this effect becomes

$$\tau c \dot{P}_c(t) + P_c(t - \tau) = F(t). \quad (60)$$

Several authors have investigated this effect in the frequency domain; however, our intent is to obtain a solution in the time domain. To accomplish this simulation, a Donner Transportation time delay electronic instrument was employed. This instrument provides a time delay by simulating the Padé approximation presented in textbooks on nonlinear systems. The general properties of the instrument are such that, if the input is $P_c(t)$, the output is $P_c(t - \tau)$. The instrument is capable of producing a large variation in τ and is an invaluable aid in analysis of this nature. Inserting the transportation delay device in the simulated loop produces an instability whose characteristics are shown in Figure V-48. In performing several runs it was found that increasing the time delay increases the amplitude of the instability but does not vary the frequency. The instability is in the form of a limit cycle. The limit cycle is a general characteristic of instability present in nonlinear systems. It was further noted that for transportation delays less than certain values the instability does not appear. These critical values and their functional relationships has not been fully evaluated at the present time.

Since the amplitude of the instability is proportional to the magnitude of the transportation time delay, the question naturally arises, what must be done to vary the frequency? It was found that for a given magnitude of transport delay the frequency could be changed by varying R/V . This was very conveniently accomplished by changing a potentiometer setting on the simulator. However, since the gas constant, R , is assumed to be constant, varying R/V may be interpreted as a variation in chamber volume, V . For a given throat area, an L^* may be defined as V/A_t , and plots of frequency versus L^* presented. Figure V-49 shows the results of this investigation.

The results of this study indicates that chamber instability is caused by the presence of a transportation delay larger than a critical value and the frequency of the instability is a function of L^* previously defined.

Although additional refinements are required to improve the dynamic behavior analyses model the effort on this contract has been completed. Further improvements to this model are being made under BAC sponsored efforts.

NOMENCLATURE

a	= Speed of sound in pipe (in/sec)
u	= Average fluid velocity along axis of pipe (in/sec)
x	= Coordinate of distance along center line of pipe (in)
p	= derivation of average pressure at a reference pipe cross section (psi)
t	= time (sec)
g	= local acceleration of gravity
K	= Fluid bulk modulus
E	= Modulus of elasticity of pipe material (psi)
f	= pipe wall thickness (in)
b	= pipe radius (in)
ρ	= fluid density (lb. per cu. in.)
y	= Valve spool displacement (in.)
K_1	= Blocked-load pressure sensitivity of valve (psi/ma)
i	= Valve command current (ma.)
K_2	= Zero load pressure droop (psi per cu in. per sec)
τ	= Droop time constant (sec)
w_n	= Apparent valve natural frequency (rad/sec)
$\mathcal{D}$	= Apparent valve damping ratio
Q	= Flow rate (cu. in. per sec.)
P	= Pressure drop (psi)
τ_c	= combustion time lag (sec)
r	= Mixture ratio
$\dot{m}$	= mass flow (slugs per sec)
A_t	= Throat area (in. ²)
p_c	= chamber pressure
c^*	= characteristic velocity (in/sec)
g	= acceleration of gravity (in/sec)

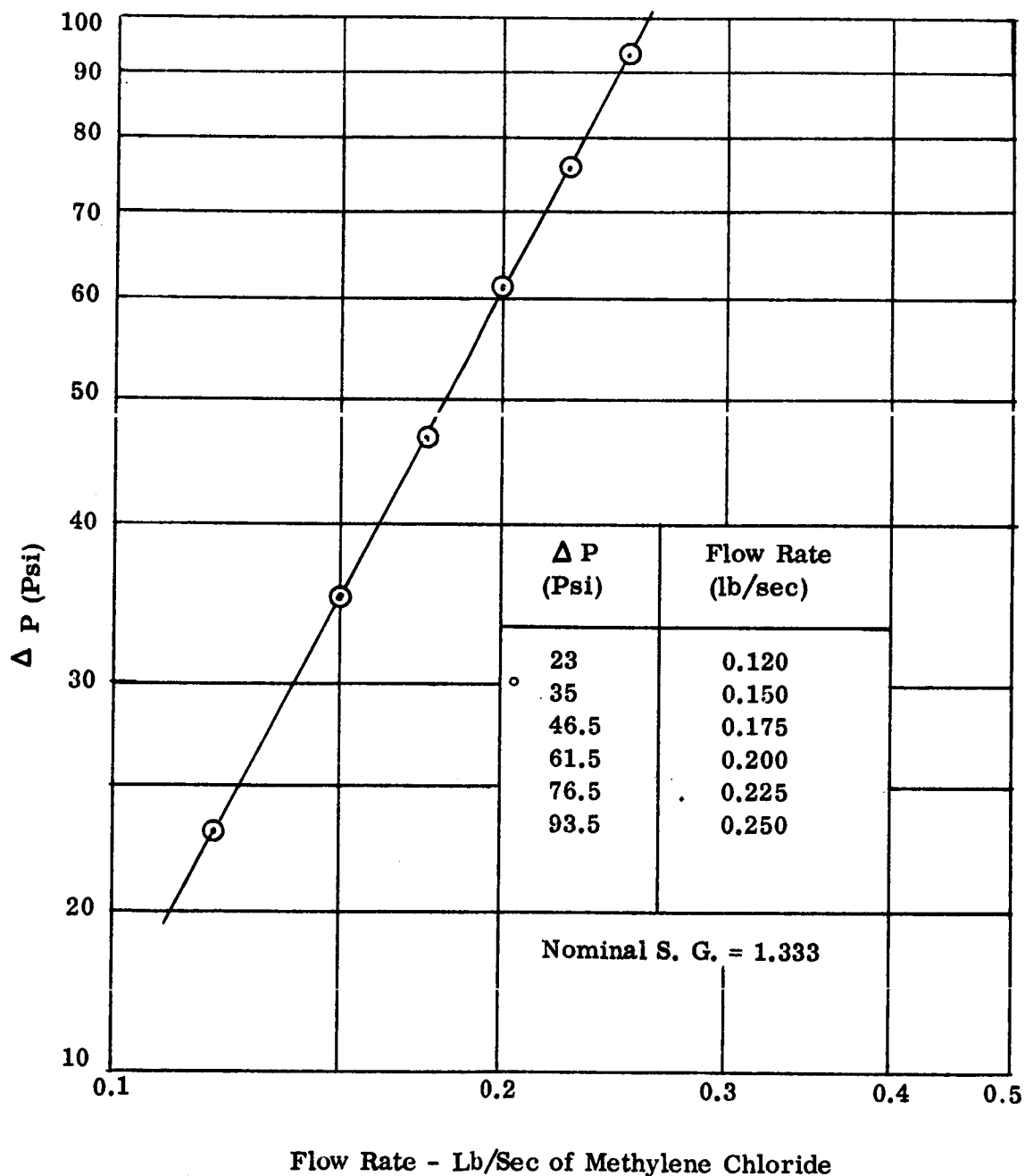


Figure V-38. Moog Bipropellant Valve Model 50-245B S/N 5 ΔP versus Flow Rate (Oxidizer Side)

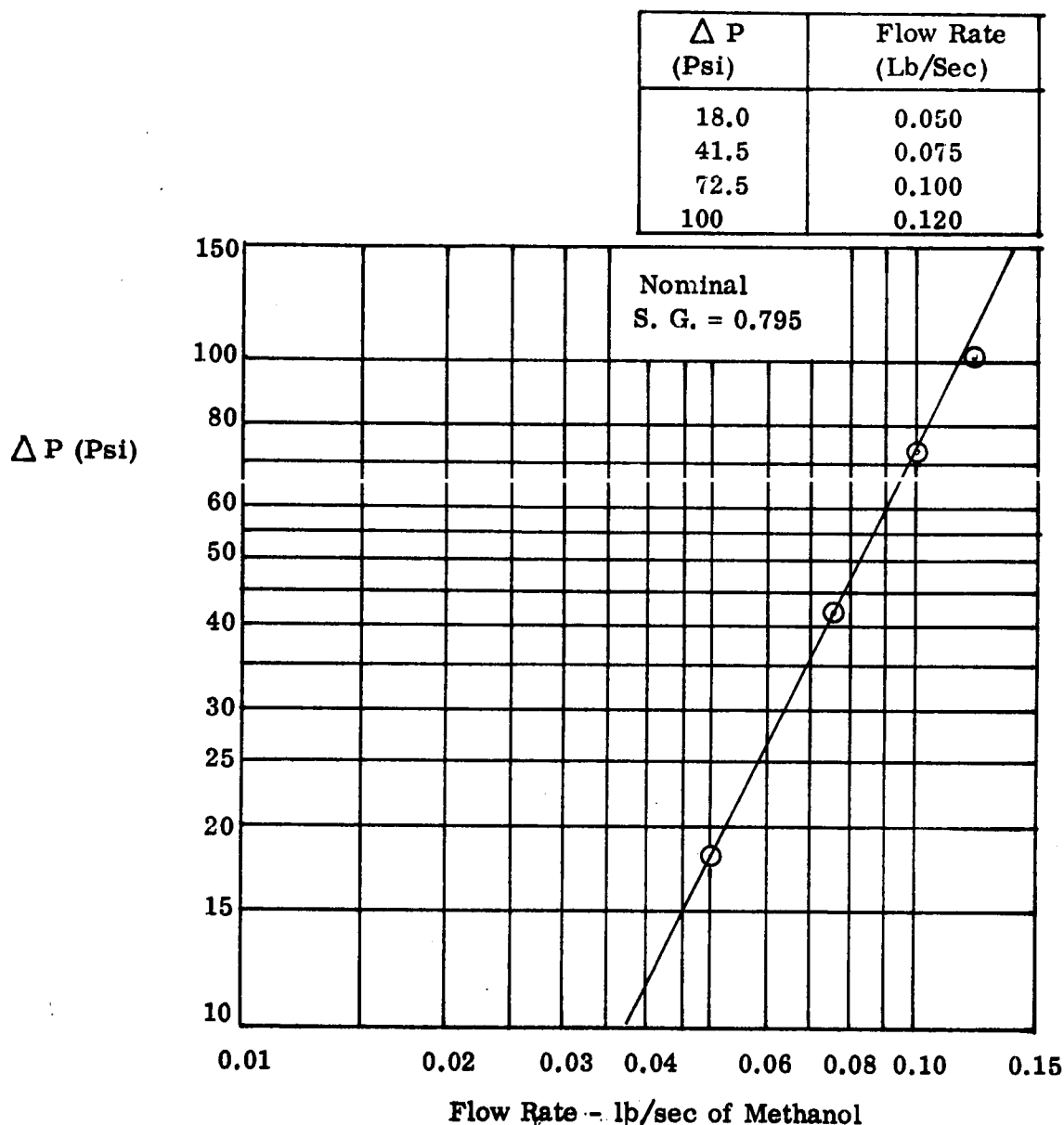


Figure V-39. Moog Bipropellant Valve Model 50-245B S/N 5 ΔP versus Flow Rate (Fuel Side)

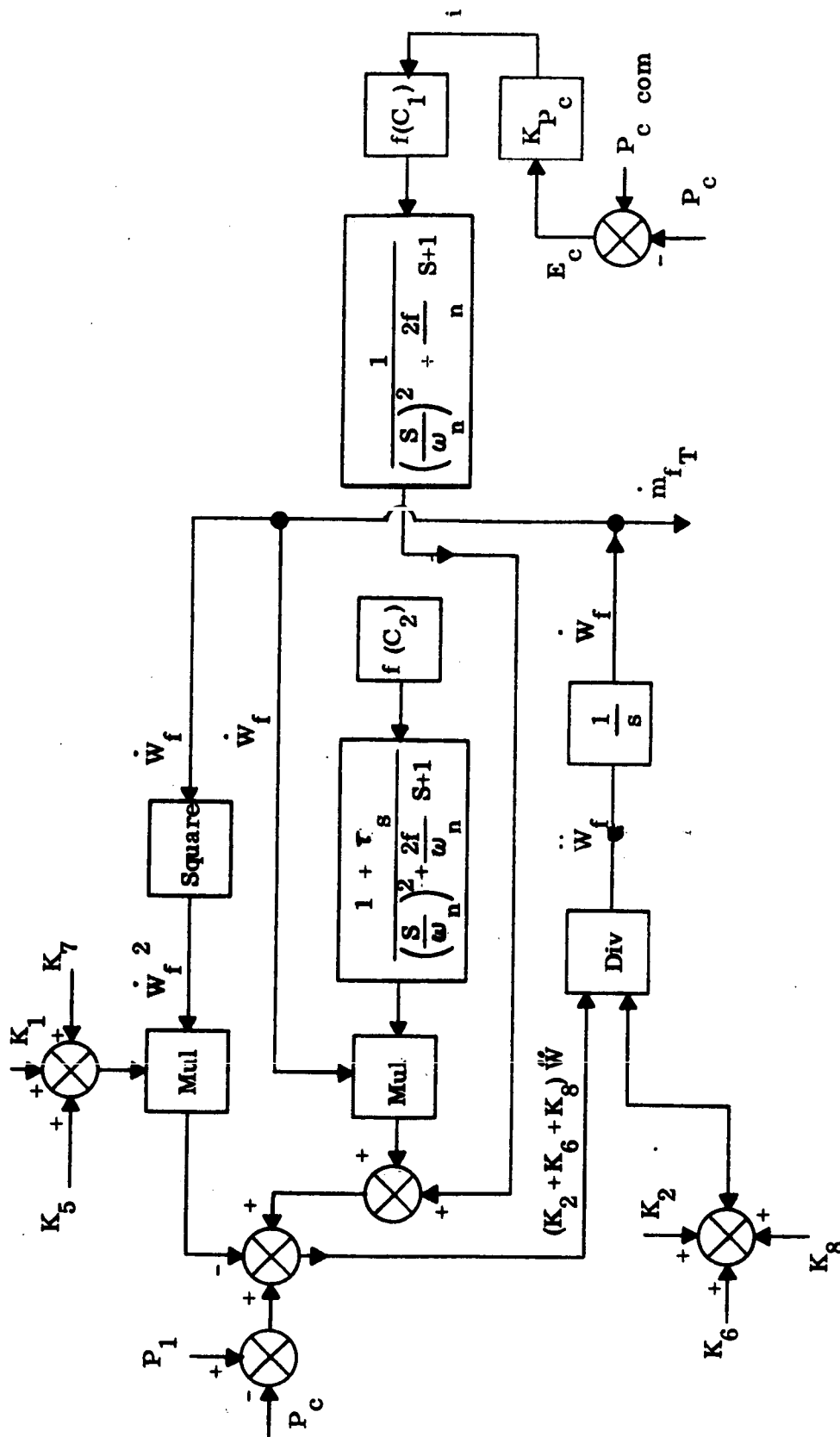


Figure V-40. Fuel Valve and Line Time Delay Dynamics

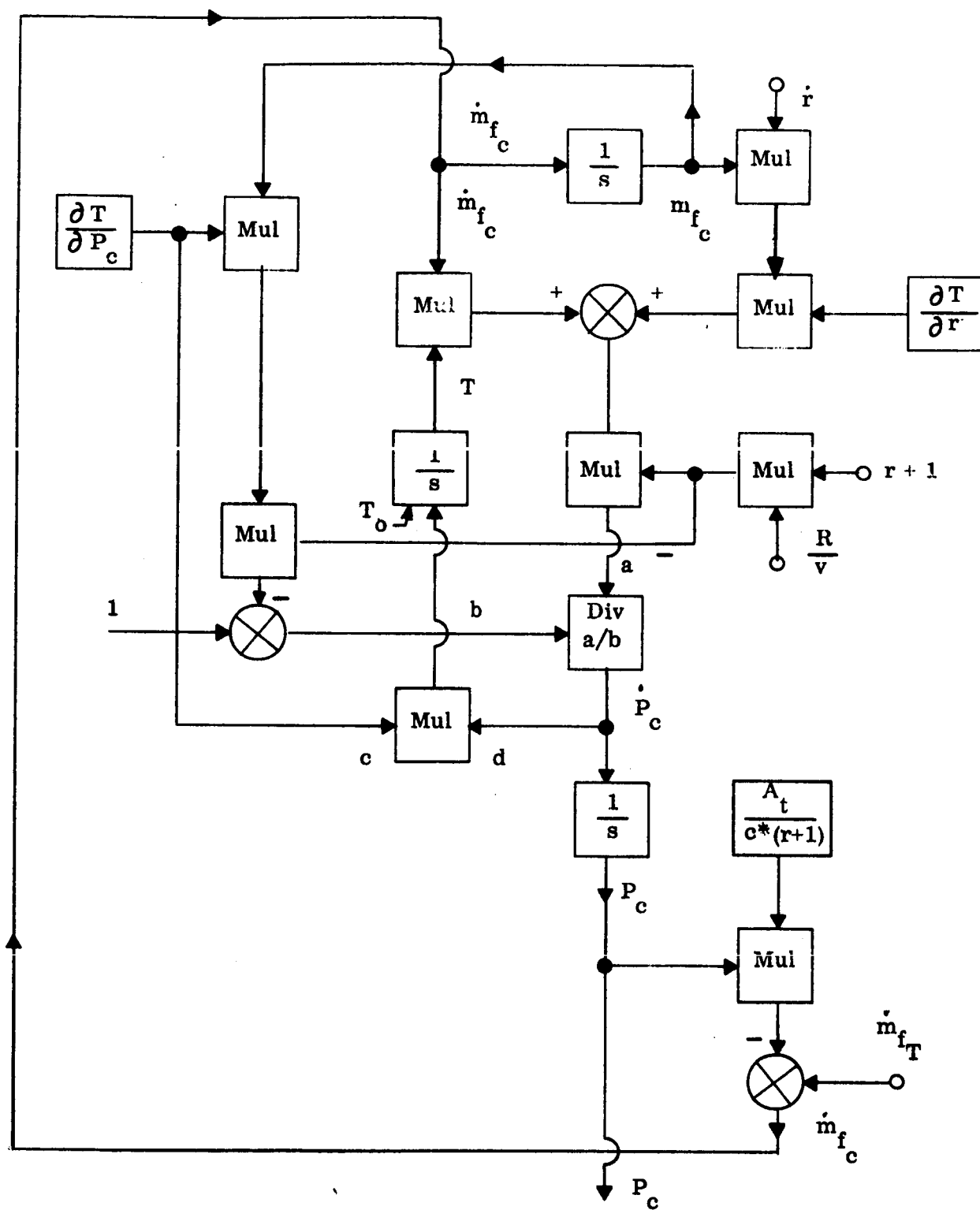


Figure V-41. Thermodynamics

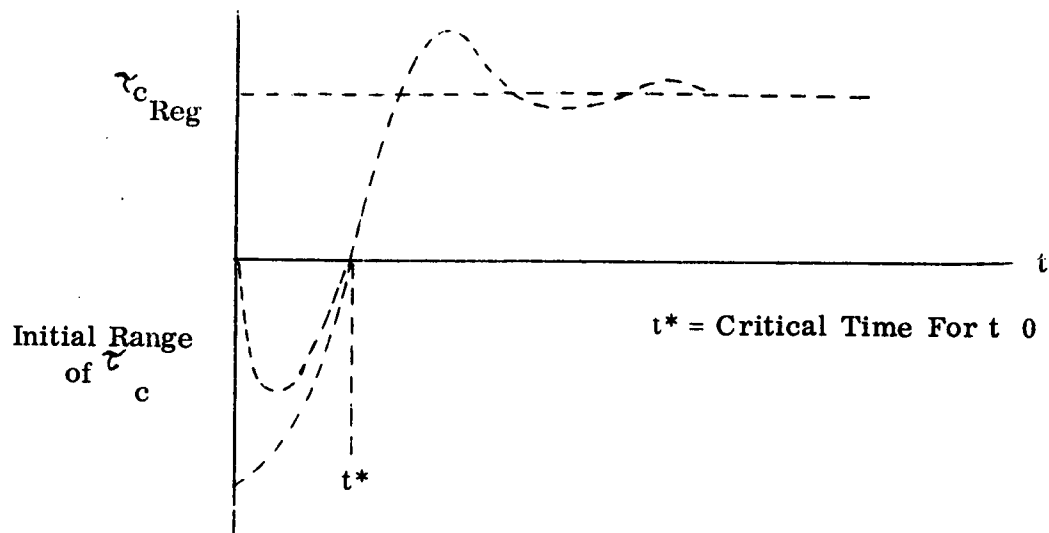


Figure V-42a. Effective Time Constant Variation

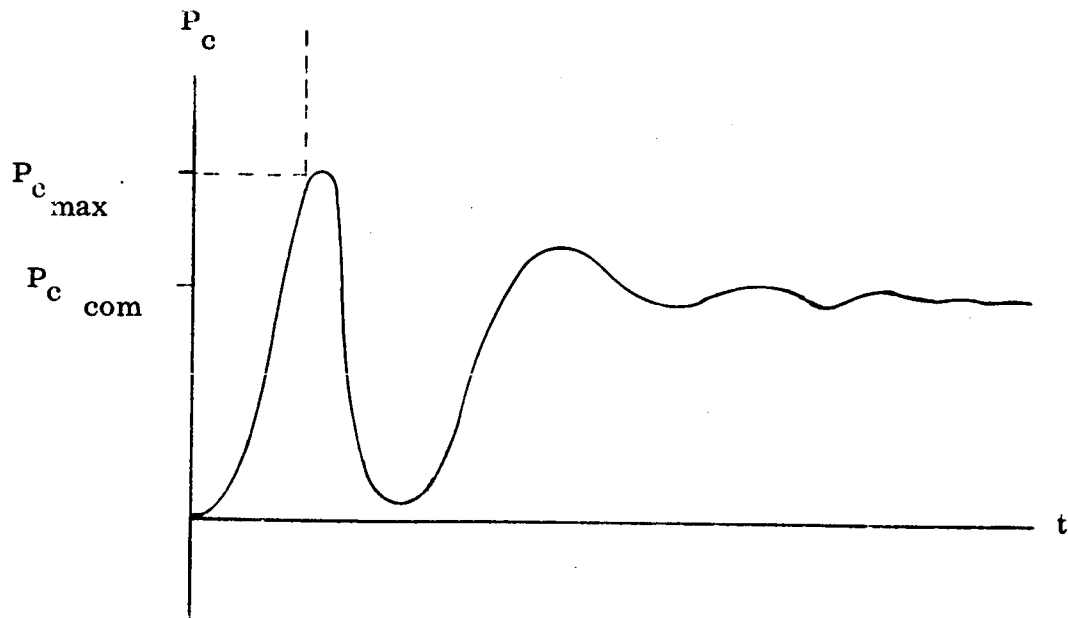


Figure V-42b. Characteristic Pressure Response

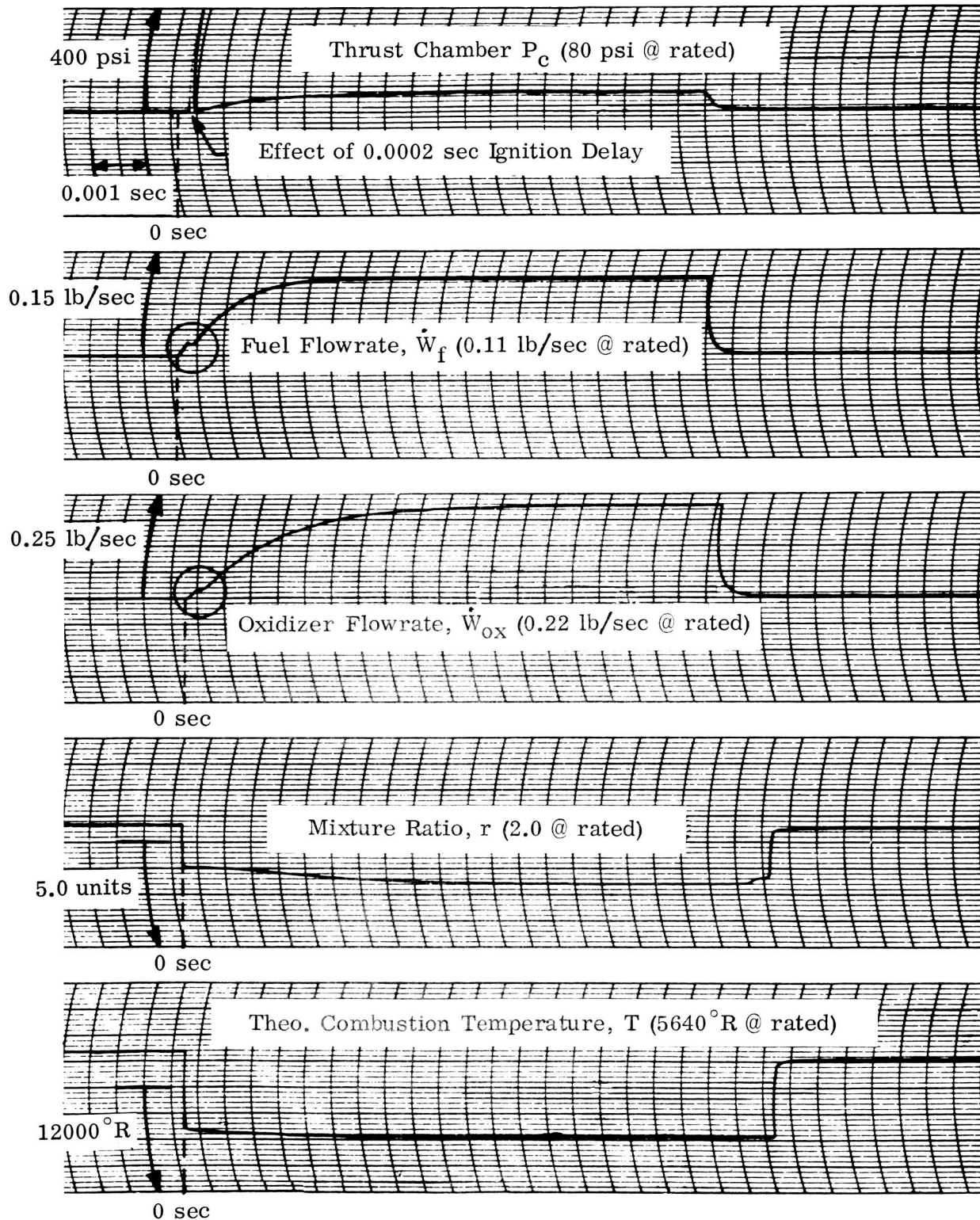


Figure V-43. Typical Analog Run - With Transport Time Delay

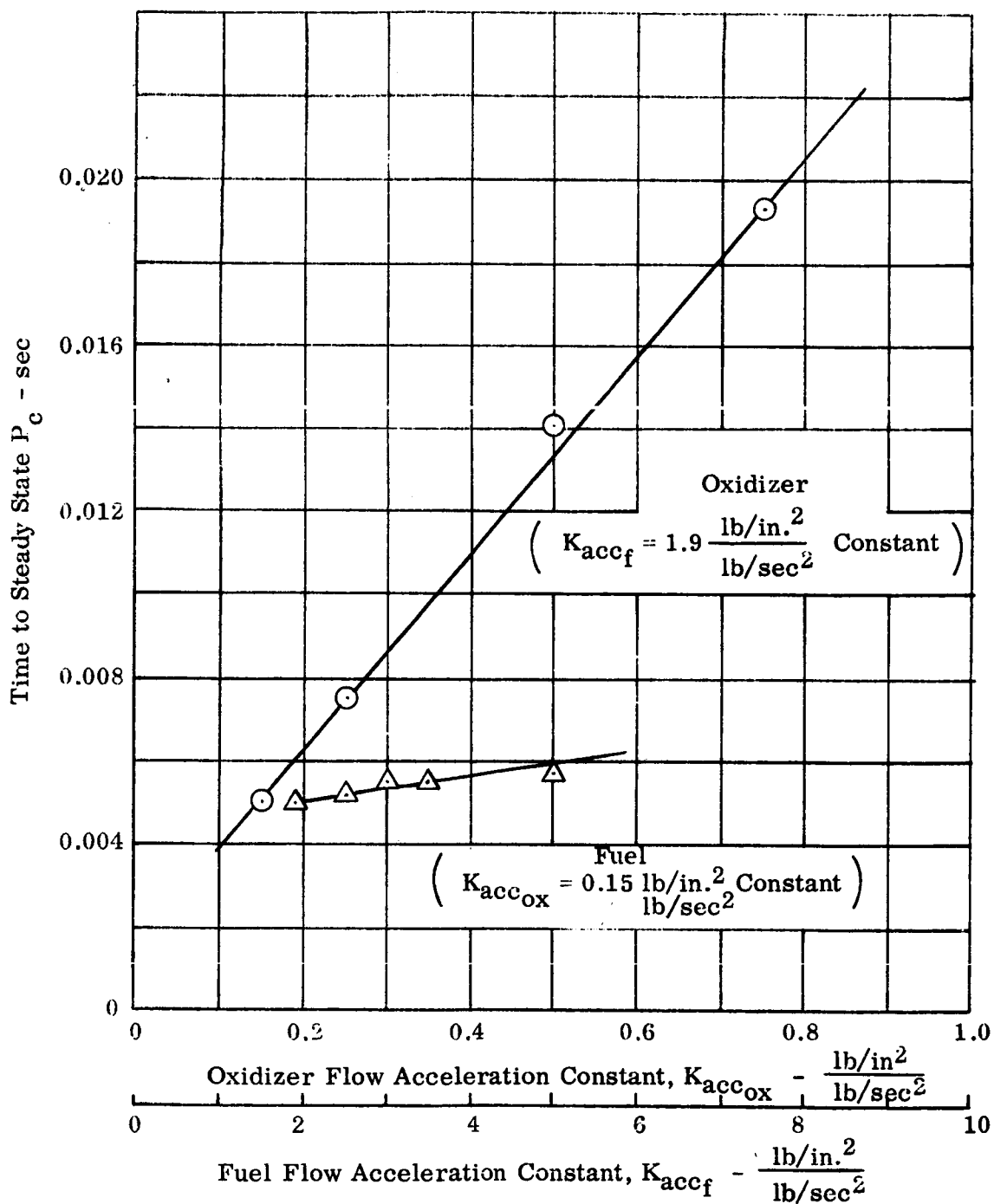


Figure V-44. Effect of Varying Propellant Acceleration Constants on Time to Steady State P_c

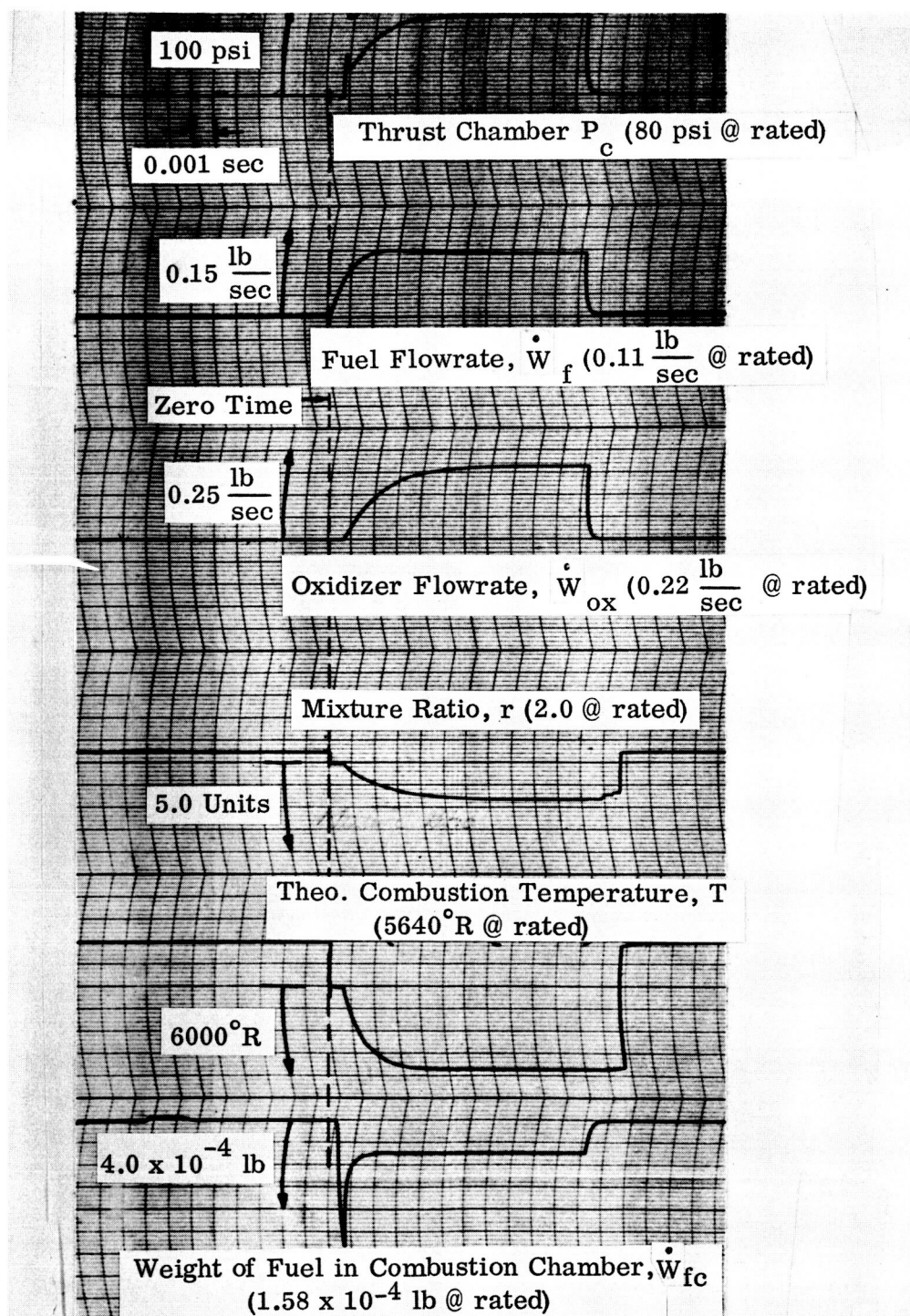


Figure V-45. Effect of Oxidizer Propellant Lag

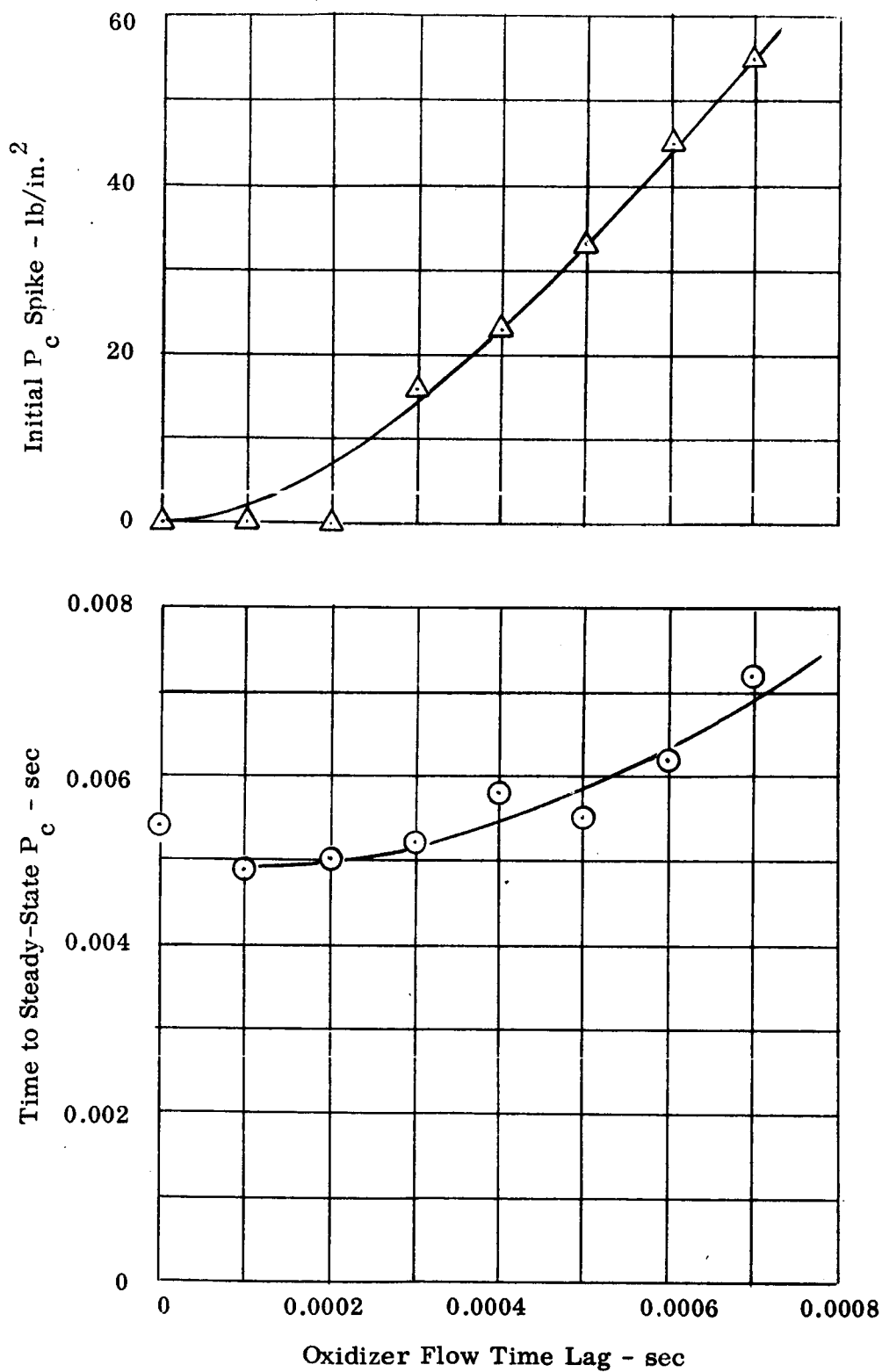


Figure V-46. Effect of Oxidizer Flow Time Lag in Reaching Thrust Chamber

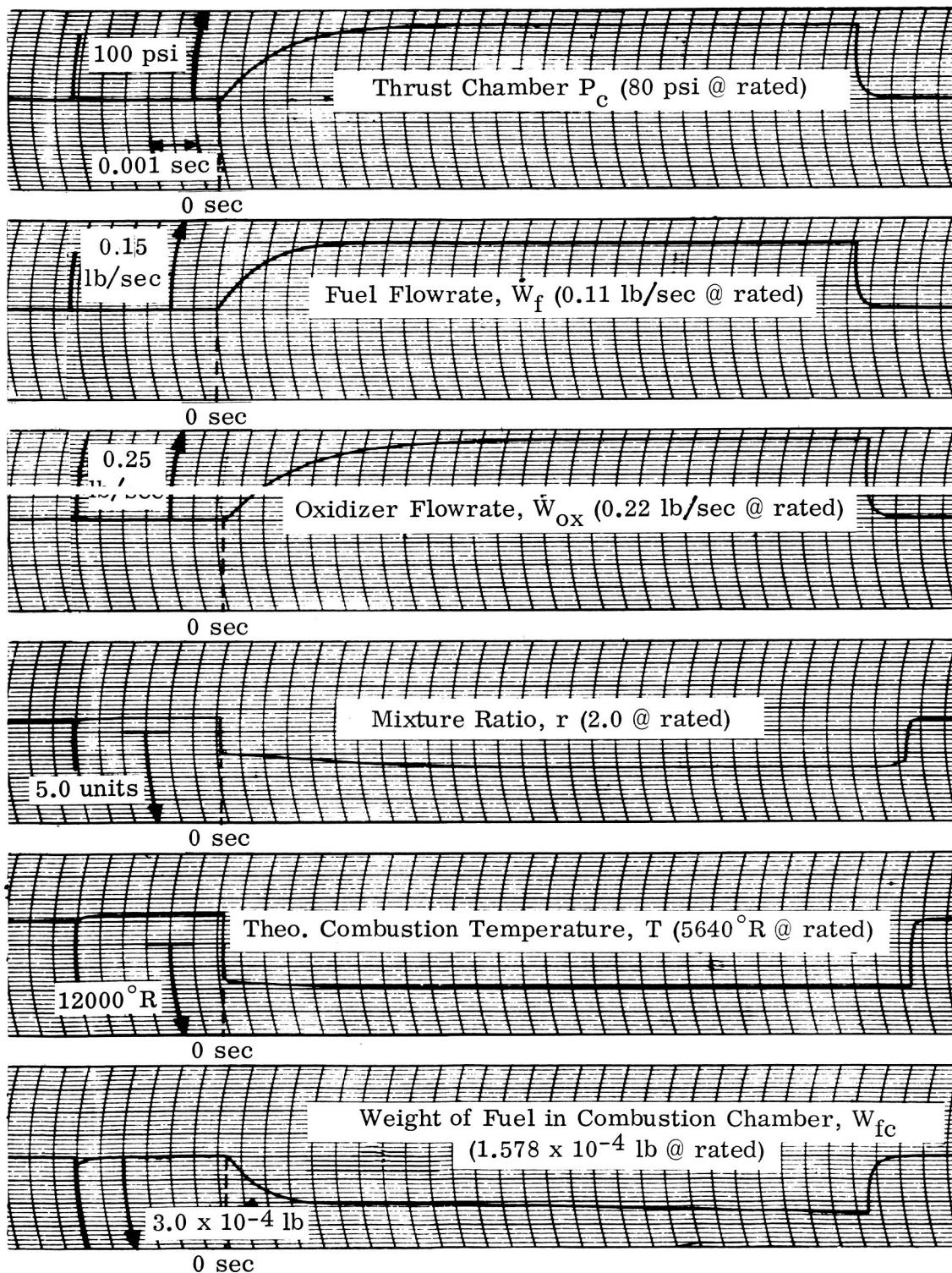


Figure V-47. Typical Analog Run - Ideal Conditions

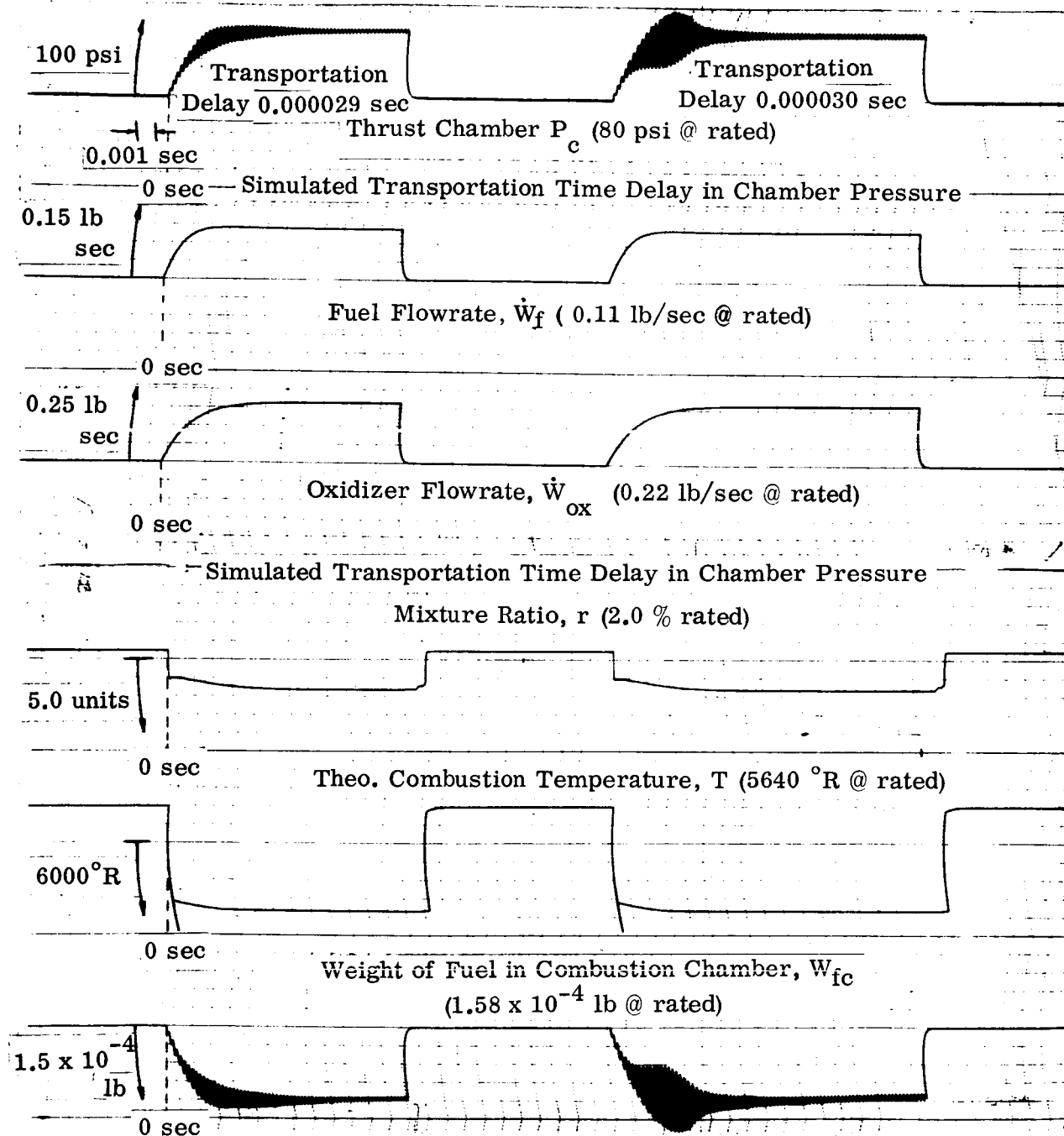


Figure V-48. Simulated Transportation Time Delay in Chamber Pressure

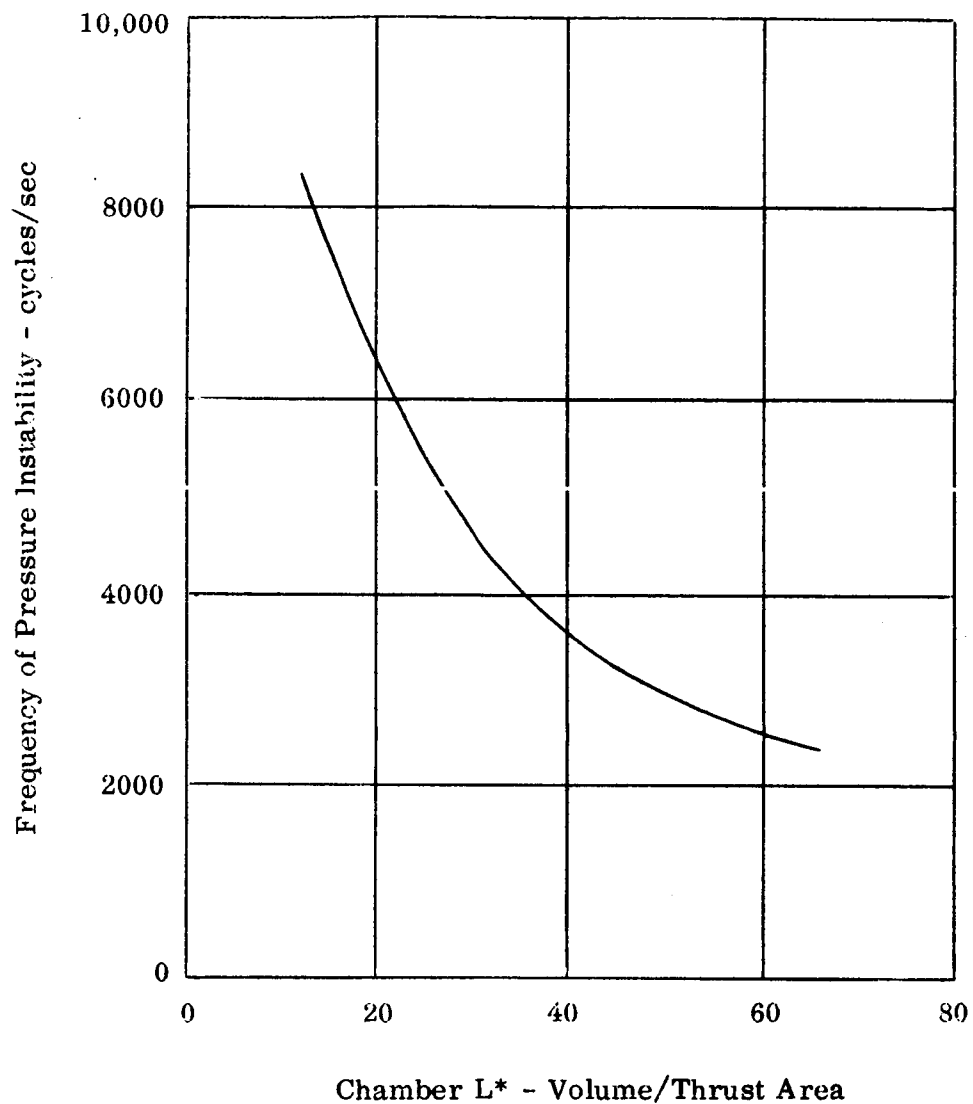


Figure V-49. Chamber L* versus Natural Frequency of Chamber P_c Due to Simulated Transportation Delay

D. INJECTOR CONCEPTS AND HARDWARE

The objective of the Task I hardware development was to support the various theoretical analyses and study efforts with actual test data. Of primary importance was the investigation of injector concepts capable of reducing the gas temperature at the chamber wall while maintaining high overall combustion efficiency.

Six injectors were evaluated during this program and are discussed in detail later in this section. To establish the heat rejection and performance characteristics of these high performance injectors, a water cooled, aluminum combustion chamber and nozzle were used initially for test purposes. Later in the program, two stainless steel chambers and two refractory metal columbium alloy chambers were used to evaluate the injector designs.

A description of each of these test chambers is given in the following paragraphs.

1. Test Hardware Design - Thrust Chambers

a. Segmented Water Cooled Injector Test Chamber

The purpose of this test chamber was to evaluate various injector configurations for:

- (1) Characteristic velocity
- (2) Total heat rejection
- (3) Circumferential heat flux distribution
- (4) Average heat flux in chamber and nozzle section

This chamber was intended to be used for sea level testing, and consequently did not include a high expansion ratio nozzle.

The design chamber pressure for this test chamber was selected to be 80 psia on the basis of a preliminary analysis which showed that it would be unlikely that the overall design objectives of the final prototype engine could be achieved at significantly higher chamber pressures. However, for test purposes the chamber can be operated up to 120 psia chamber pressure.

The chamber internal geometry was determined predominantly by establishing a minimum diameter, compatible with the projected minimum injector diameters. Some compromise had to be made on the chamber diameter to avoid an unduly long chamber which is ultimately undesirable from engine structural and envelope considerations. To minimize potential hot gas recirculation on the injector end, the head end was slightly tapered to meet the minimum injector face diameter.

This minimum injector face diameter had been independently established as a compromise between injector face propellant loading, minimum heat transfer area and mechanical design considerations in accommodating properly the necessary injection elements.

In order to accommodate, in this test chamber, all injector variations, including a film cooled type which requires a larger diameter in the area of the film cooling orifices, the head end of the test chamber was scalloped in 8 locations.

The test chamber design parameters are listed below:

Chamber Pressure	80 psia Nominal
Expansion Ratio	2.0
Contraction Ratio	3.87
Characteristic Length	15 inches
Throat Area	0.694 inch ²
Coolant Flow Rate, H ₂ O	0.418 Nominal
Coolant Temperature Rise, Chamber	58°F
Coolant Temperature Rise, Nozzle	41°F
Coolant Pressure Drop	Est. 8 psia
Coolant Velocity, Chamber	13 ft/sec
Coolant Velocity, Throat	20 ft/sec

Figure V-50 shows the assembly drawing of this chamber and photographs are given in Figures V-51 through V-53. In the first photograph, an exploded view of the chamber shows the milled water passages in the aluminum nozzle and barrel sections. The second photograph (Figure V-52) shows the assembled chamber without the thermocouple probes installed. In Figure V-53, the completed water-cooled chamber assembly is shown with the thermocouple probes in position. The inlet water temperature at the nozzle is monitored in

each of the two water inlet feed tubes. The water temperatures at the nozzle water outlet station (at the entrance to the nozzle convergence) are measured in 16 locations. Each of these temperature measurements is an average of three nozzle water passages which are manifolded together in the vicinity of the temperature probe. The nozzle outlet temperatures also represent the chamber (barrel section) water inlet temperatures.

The water in the chamber section flows back and forth through three passages in series to achieve a higher water temperature rise thus providing more accurate temperature differential measurements. At the injector end of the chamber, the water temperature is again measured in each of the 16 barrel sectors prior to discharge from the chamber. The drawing in Figure V-54 represents a typical test setup with the water-cooled chamber. The thermocouple arrangements and locations are indicated on this drawing. All water temperature measurements were made with chromel-alumel thermocouples.

Prior to utilizing the water-cooled test chamber in fire tests, a water flow calibration was made to determine the pressure drop and the percent of water flow through each of the 16 chamber segments. The pressure drop for various total water flow rates through the chamber is presented in Figure V-55. The typical water flow distribution for the test chamber is tabulated below:

Flow Distribution at a Total Flow Rate of 0.418 Lb-H₂O/Sec
T.C.A. S/N 1

<u>Segment No.</u>	<u>% of Total Flow Through Each Segment</u>	<u>Segment No.</u>	<u>% of Total Flow Through Each Segment</u>
1	6.36	9	7.05
2	5.50	10	7.31
3	4.82	11	6.84
4	4.90	12	6.75
5	4.90	13	7.05
6	5.67	14	6.71
7	6.15	15	6.45
8	6.79	16	6.75
			100.00%

b. Stainless Steel Start Chambers

Two stainless steel chambers were fabricated for use with the above mentioned injectors. The first chamber was fabricated as part of the NAS 7-305 contract, the second chamber was fabricated on BAC sponsored funds. The throat areas for these chambers were sized to obtain chamber pressures of 80 and 135 psia respectively, for operation with a constant injector propellant weight flow. In this manner, a given injector could be operated in either thrust chamber without changing the propellant injection pattern at the nominal design operating condition. Further, a constant chamber characteristic length (L^*) of 15 inches was maintained in the design of both units.

The stainless steel chambers were utilized on this program to:

- (1) Provide preliminary estimates of hot wall chamber heat rejection characteristics at 80 and 135 psia chamber pressure operation.
- (2) Obtain comparative performance (c^*) figures at both chamber pressure values.
- (3) Conduct vacuum ignition pulse tests using high frequency response instrumentation.

The 80 psia chamber pressure steel chamber is shown in Figure V-56 and is representative of both units. The flange attached to the nozzle end of the chamber (Figure V-57) provides the interface between the thrust unit and the vacuum tank used for high altitude ignition tests.

c. Hot Wall Columbium Test Chambers

To better correlate heat rejection rates obtained with the water cooled test chamber to hot wall heat rejection rates and to study in more detail injector effects on the hot wall and coating, two columbium test chambers were used. Both chambers were of identical design except for the method of attaching thermocouples. The first columbium chamber design incorporated weld beads around the periphery of the chamber midway between the injector end and the convergent nozzle. These weld beads were necessary due to the thin wall in the chamber section. Thermocouples were attached to this chamber section

by means of drilling a hole in each weld bead, inserting the thermocouples and peening each thermocouple in place. Thermocouples at the throat station were attached in a similar fashion by drilling holes into the thick wall at the throat then peened in place. The second columbium chamber incorporated three rings around the chamber section giving more versatility to the location of thermocouples. As with the first chamber, thermocouples were attached by drilling hole in the three rings and at the throat station, inserting the thermocouples and peening them in place. These chambers, illustrated in Figures V-58 through V-60, were fabricated from a columbium-1% zirconium alloy. A silicide coating (Sylcor R508C) was applied to both internal and external surfaces to provide the columbium alloy base metal with high temperature oxidation resistance. The internal geometry of each chamber is identical to that of the water-cooled test chamber and the stainless steel chamber.

2. Test Hardware Design - Injectors

Based on the previous and successful experience with the six element triplet injector, variations to this basic design were made during this program. The main objective of the entire injector study was to develop a means of reducing the gas temperature at the chamber wall while obtaining high combustion efficiency. The goal was to develop an injector delivering a characteristic velocity of 5350 ft/sec (equivalent to a specific impulse of 290 seconds at an area ratio of 40:1) and a driving gas temperature below 3000°F. Since the basic six element triplet injector design met this performance level with an estimated driving gas temperature of 3500°F, a decrease of 500°F in driving gas temperature was needed.

The state-of-the-art chamber and coating systems are presently limited to approximately 3000°F for the long term usage of one hour. With an injector generating a driving gas temperature below 3000°F, a chamber constructed from these materials and coatings can then be insulated since, in the nearly adiabatic condition, the chamber wall temperature approaches the driving gas temperature.

Following is a description of the first four of six injector designs evaluated and the testing results.

All test injectors were simplified versions of a finally contemplated flight type, fabricated from stainless steel. However, great care was taken to design all flow passages, residual volumes, injector valve interface and other hydraulic characteristics as near to the envisioned final configuration as could be determined.

a. Injectors T-1 through TS-4

(1) Injector Description, T-1 through TS-4

(a) Injector T-1

This injector is a balanced eight element design where each element consists of two symmetrical fuel streams impinging on a central oxidizer stream. Injector T-1 is a direct derivative of the Bell 100-lb thrust injector Model 8290. The number of injection elements has been increased from 6 to 8 in order to achieve more uniform circumferential propellant distribution. Injection parameters were chosen to match those where the Model 8290 injector produced best performance and lowest wall temperatures. Adjustments in orifice diameters to compensate for the change in propellants from N_2O_4 - 50% N_2H_4 50% UDMH to N_2O_4 - MMH and change in design mixture ratio from 2.0 to 1.60 were made. The injector design is shown in Figures V-61 and V-62.

(b) Injector TF-2

Injector TF-2 represents a triplet type with 8 balanced triplets in the core, surrounded by eight discrete film cooling orifices, spaced in between the triplet elements. The film cooling orifices are of a single orifice splash plate type, designed to inject a narrow fuel fan along the wall. These orifices are sized to inject 10% of the rated fuel flow along the thrust chamber wall. However on this test injector, these film cooling orifices are supplied separately from the main fuel orifices such that the film cooling rate may be changed during test or cut in and out to accurately determine the effects of the film cooling arrangement on performance and wall temperatures. This design is shown in Figures V-63 and V-64.

(c) Injector TU-3

The third variation in triplet element injector design is test injector TU-3. This type of triplet element again contains two fuel streams impinging on a centrally located axial oxidizer jet. The element is classified as unbalanced, however, since the design forces a greater mass flow of fuel through one of the fuel orifices in comparison to the matching fuel orifice. The unbalanced fuel mass flow condition yields a nonaxial resultant momentum vector following stream impingement. To compensate for this, it was necessary to vary the fuel orifice injection angles to again achieve the desired axial resultant momentum condition. This design is shown in Figure V-65 and V-66.

The unbalanced triplet injector design is an attempt to provide a fuel rich, relatively low temperature gas mixture in the vicinity of the thrust chamber wall. Further, it is an attempt to accomplish this gradation of propellant mixture ratio across each triplet element of the basic injector thereby eliminating the need for a secondary cooling mechanism.

Figures V-67 and V-68 illustrate a water flow model of an unbalanced triplet element. The injector element is operating at an unbalanced flow condition, where there is approximately one-third more mass flow through one of the fuel orifices. Injection velocities for this flow condition are 36 and 43 feet per second for the fuel and oxidizer respectively. This particular water flow model with the flow conditions listed, served as the basis for test injector TU-3. The design selection was based upon colored water flow tests in which local mixture ratio conditions across the spray fan were evaluated by color index. This method serves only as a coarse and qualitative screening process. It is, however, a useful tool for making preliminary design selections in the absence of previous fire test data for a given injector design.

(d) Injector TS-4

Injector TS-4 is termed a "split triplet" and is pictured in Figures V-69 and V-70. The split triplet is a further attempt to physically effect a controlled mixture ratio gradation of propellants within individual injector elements to the extent that a fuel rich mixture ratio zone is created adjacent to the chamber wall.

A conventional triplet pattern (fuel-on-oxidizer elements) was divided into two parts by splitting the oxidizer jet into two streams. As can be seen from the water flow model pictures (Figures V-71 and V-72), the oxidizer flow remains centrally located in two axial injection streams. The two impinging fuel streams are paired with the two oxidizer streams initially forming two doublets. The resultant momentum of each doublet is unbalanced and as a result, secondary fan impingement takes place. The resultant momentum of this secondary impingement is predominantly axial thus preserving the desired characteristic of a triplet design.

By physically splitting the triplet element, the mixture ratio delivered through each half of the triplet can be controlled and varied. In other words, that half of the split triplet located in the center portion of the injector face can be run oxidizer rich producing a high performing hot core of combustion gases. Conversely, the outer half of the split triplet can be operated fuel rich to provide a shroud of cooler gases between the core and the chamber wall. In addition, the amount of mass flow through each half of the split triplet can be unbalanced. A greater proportion of the total mass flow can be directed toward the relatively cool burning (fuel rich) zone along the chamber wall to effect reductions in the driving gas temperature.

The determination of optimum design parameters for a split triplet configuration can only be achieved empirically through a logical sequence of design and test verification. To initiate a design for performance and heat rejection evaluation, it was decided to drive fifty percent of the propellant flow (per element) through each half of the split triplet. The outer half of the element was designed for a nominal mixture ratio of 1.4. To achieve an overall mixture ratio of 1.6, the inner half of the split triplet was sized for a 1.8 mixture ratio.

(2) Injector Test Results - Water Cooled Chamber

The four test injector configurations described above were fire tested in the water cooled thrust chamber for a total of 47 tests. The photograph in Figure V-73 shows a typical example of an injector assembled to this chamber.

A total of nine tests were completed on injector T-1, 13 tests on injector TF-2 with film coolant and five tests without film coolant, 10 tests on injector TU-3, and 10 tests on injector TS-4.

A summary of these tests is given in Table V-8. Performance curves for the various injectors are illustrated in Figure V-74. Injector TS-4 consistently exhibited the highest performance (94.8% of shifting theoretical c^* at a 1.6 mixture ratio) for all mixture ratio conditions. As such, it serves as a performance reference for the other injectors tested. Test Injector T-1 was originally intended to serve as a performance standard, however, manufacturing imperfections compromised the impingement spray pattern. As a result, the performance of injector T-1 was substantially the same as that of injector TF-2 employing 13% of the fuel flow for film cooling purposes. Injector TF-2 with 13% film cooling produced a performance curve which peaked at a mixture ratio between 1.4 and 1.5. When the same injector was fired without film cooling (total fuel flow injected through the primary fuel orifices), the peak performance mixture ratio shifted to approximately 1.7.

In general, performance was directly proportional to heat rejection. This is shown in Figure V-75 where average values for total heat rejection rates are plotted against measured characteristic velocities for the various injectors. In addition to the performance levels reported in Figure V-74 for injector TF-2 at 0 and 13% film cooling the limited testing conducted at film cooling rates of 7, 10, and 22% are also shown in Figure V-75.

To complete the relationship of heat rejection rates to performance, a theoretical line was drawn in Figure V-75 based upon the Eckert Enthalpy method. This theoretical curve serves as an arbitrary reference to the actual data. If a line were drawn through the data points for injectors TU-3, T-1 and T-4, the slope of such a line would approximate that of the theoretical reference. This tends to confirm that heat rejection is highly sensitive to the absolute performance level of a given injector. One way to alter this condition significantly is to utilize film cooling. In this manner, a positive fuel rich barrier condition may be established at the thrust chamber wall. The line drawn through the TF-2

TABLE V-8
WATER COOLED CHAMBER DATA SUMMARY

Injector S/N	Run No.	Run Duration in sec.	Data at "t" sec	Film Cooled Flow %	P _c psia	O/F Ratio	Measured c*-ft/sec	η_c %	Average Total Q/A BTU/sec/in. ²
T-1	2083	10.4	9.5	-	80.6	1.54	528.4	92.8	1.17
	2084	10.2	9.5	-	79.7	1.59	525.9	92.3	1.15
	2085	20.5	19.0	-	79.6	1.59	525.7	92.3	1.20
	2086	20.9	19.0	-	79.3	1.59	523.9	92.0	1.19
	2087	20.7	19.0	-	76.2	1.72	518.5	91.0	1.02
	2088	20.8	19.0	-	75.9	1.30	523.2	93.2	1.19
	2089	21.3	19.0	-	65.8	1.48	528.0	93.0	0.97
	2090	20.8	19.0	-	79.3	1.56	526.4	92.5	1.18
	2091	20.5	19.0	-	86.7	1.55	524.2	92.1	1.32
	1987	9.9	9.5	13	70.6	1.29	524.6	93.6	0.84
	1988	14.1	14.0	13	78.9	1.55	525.2	92.4	0.82
TF-2	1989	19.1	19.0	10	79.2	1.53	528.8	92.8	1.01
	1990	19.1	19.0	7.3	79.2	1.54	530.3	92.8	1.15
	1991	19.1	19.0	22.2	75.2	1.49	509.9	89.9	0.73
	1992	19.1	19.0	13	78.5	1.53	525.6	92.5	0.93
	1993	19.1	19.0	13	77.4	1.72	519.4	91.5	0.86
	1994	19.1	19.0	13	58.8	1.52	521.7	91.8	0.63
	1995	19.1	19.0	13	84.0	1.49	526.9	92.8	1.04
	1996	19.1	19.0	13	72.8	1.69	520.5	91.5	0.81
	1997	19.1	19.0	13	90.6	1.62	522.8	91.8	1.11
	1998	20.4	19.9	13	81.2	1.42	522.0	93.1	1.00
	1999	20.2	19.7	13	81.9	1.66	521.5	91.4	0.94
	2078	10.6	9.5	0	77.3	1.57	522.3	91.7	1.15
	2079	30.6	29.0	0	77.5	1.59	533.6	93.4	1.19
	2080	20.5	19.0	0	77.6	1.59	533.6	93.7	1.17
	2081	20.6	19.0	0	78.0	1.74	534.7	93.9	1.15
	2082	20.6	19.0	0	76.5	1.38	520.4	92.0	1.17

TABLE V-8 (CONT)

Injector S/N	Run No.	Run Duration in sec.	Data at "t" sec	Film Cooled Flow %	P _c psia	O/F Ratio	Measured c*-ft./sec	η_c %	Average Total Q/A BTU/sec/in. ²
TU-3	2034	10.3	9.5	-	75.2	1.51	4976	87.5	0.81
	2035	10.8	9.5	-	78.2	1.64	4919	86.3	0.81
	2036	10.7	9.5	-	78.9	1.68	4889	85.8	0.84
	2037	30.8	29.0	-	78.9	1.69	4868	85.5	0.83
	2038	30.8	29.0	-	77.5	1.63	4887	85.8	0.93
	2039	30.6	29.0	-	76.6	1.40	4968	87.7	0.96
	2040	31.3	29.0	-	74.1	1.75	4836	85.0	0.76
	2041	31.0	29.0	-	65.7	1.57	4886	85.9	0.64
	2042	30.9	29.0	-	84.5	1.55	4922	86.5	0.86
	2043	31.1	29.0	-	76.6	1.62	4871	85.5	0.79
TS-4	2017	5.7	5.2	-	82.8	1.54	5397	94.9	1.35
	2018	30.5	29	-	82.9	1.56	5390	94.8	1.39
	2019	30.5	29	-	82.7	1.56	5385	94.6	1.33
	2020	30.5	29	-	74.2	1.61	5397	94.8	1.25
	2021	30.4	29	-	83.0	1.76	5369	94.2	1.34
	2022	16.6	14	-	82.7	1.39	5331	95.3	1.41
	2023	TEST STAND MALFUNCTION							
	2024	30.5	29	-	81.6	1.50	5393	95.0	1.44
	2025	30.3	29	-	91.0	1.50	5382	94.8	1.54
	2026	30.7	29	-	79.9	1.32	5296	94.2	1.36

injector data points (Figure V-75) for various film coolant flows illustrates this point. A significant change in the slope of the line occurred in comparison with the theoretical reference. Stated another way, the film cooled injector exhibited a greater reduction in heat rejection rates for a correspondingly smaller loss in performance by comparison to the other test injectors or the theoretical reference. The results indicate, however, that the film cooling is not very effective at the throat station.

To achieve adiabatic wall operation at a performance level of 290 seconds specific impulse, heat rejection rates must be limited to approximately 0.9 BTU/sec/in.². This is also shown in Figure V-75 in relation to the test results from the first four injector designs.

The values of Q/A for injector TF-2 at various percent film coolant flows are included in the Appendix to this report in the form of a Cartesian and polar coordinate plots. A sample of the type of data processing used for injector TS-4 tests is also included in the Appendix.

(3) Injector Test Results - Stainless Steel Chamber

Injector TF-2 (film-cooled) was assembled to a stainless steel start chamber designed for a nominal chamber pressure of 80 psia. A series of performance runs were made with a 13 percent film coolant condition and a nominal chamber pressure of 80 psia. The last five seconds of the last test were conducted at zero percent film coolant. This test sequence was repeated with a stainless steel chamber sized for a 135 psia chamber pressure. A second run series was made using the high chamber pressure start chamber with the percent of film coolant at 13 and 22 percent.

These runs were made to obtain preliminary information regarding film cooled injector performance and heat rejection characteristics in a hot wall chamber. Heat rejection, in this case, could only be qualitatively observed by monitoring wall temperatures and temperature change rates up to 1900°F. The data obtained from these runs are tabulated in Table V-9. Performance curves for these runs are illustrated in Figures V-76 and V-77.

TABLE V-9
INJECTOR TF-2/STAINLESS STEEL CHAMBER DATA

80 psia Thrust Chamber
(Runs 1982 - 1986)

Run No.	Run Duration in Sec	Data Pt at "t" Sec	% Film Coolant	P _c psia	O/F Ratio	Measured c*-Ft/sec	η_{c^*} %	Max. Wall Temp-°F
1982	5.3	4.8	13	78.3	1.53	5323	93.6	1152
1983	6.9	6.4	13	77.7	1.53	5330	93.7	1212
1984	10.4	9.9	13	76.1	1.76	5234	91.8	1319
1985	19.5	19.0	13	78.3	1.62	5268	92.6	1848
1986	10.0	4.4	13	79.4	1.59	5328	93.5	1245
		9.5	0	78.3	1.76	5438	95.6	

135 psia Thrust Chamber
(Runs 2009 - 2016)

2009	5.1	Instrumentation Malfunction - Lost Oxidizer Flow						
2010	6.5	Film Coolant Valve Malfunction						
2011	5.5	5.0	22	128	1.27	5275	94.3	685
2012	7.0	6.5	22	126	1.74	5010	87.8	774
2013	20.1	19.6	22	130	1.55	5140	90.3	1833
2014	5.7	5.2	13	136	1.35	5310	94.1	1049
2015	6.4	5.9	13	135	1.72	5280	92.7	1065
2016	12.3	11.8	13	134	1.55	5280	92.7	1592

(4) Determination of Combustion Gas Temperature by Reduction of Test Data

The pacing item for classes of materials, both metallic wall and insulation, has been the combustion temperature of actual propellants adjacent to the wall. Several ways for determining the gas temperature are possible. The most exact method is the measurement of an adiabatic wall with insulation since the wall temperature is the combustion temperature. Hot wall insulated tests will be discussed later in this section. Initially, however the tests results of injector TF-2 in the stainless steel chambers and the water cooled chamber were analyzed.

(a) Stainless Steel Thrust Chamber Transient Data

The series of film cooling injector tests were analyzed for estimates of the driving gas temperature.

Thermocouples at three axial locations sensed the temperatures at the outside wall with four thermocouples at each station. Their location is shown in Figure V-78. Run durations of 20 seconds were made due to the material limitations of the stainless steel, approximately 1900°F.

The measured test data for the outside wall temperature as a function of time are used to determine the gas temperature at the wall and the heat transfer coefficient during the transient by establishing a heat balance. At any instant, the heat added to an element of the chamber is equated to the time rate of temperature rise of a thermal mass plus the radiant heat lost by the outer wall to its environment.

The following equation is solved at two instants in one location for each firing:

$$\dot{q} = h_g \left[T_g - T_{wg} \right] = \frac{A_2}{A_1} \left[\sigma \epsilon T_{wg}^4 + \text{thermal mass} \frac{dT_{wg}}{d\theta} \right]$$

where

$\dot{q}$ = heat flux, B/in.² sec

h_g = gas film coefficient, B/in.² sec °F

T_g = gas temperature at wall, °R

T_{wg} = wall temperature, °R

σ = Stefan-Boltzmann constant

ϵ = outer emissivity

$\frac{dT_{wg}}{d\theta}$ = time rate of wall temperature change, °F/sec

The two instants considered were $\theta \approx 4$ seconds and near the end of the firing. Typical transients are shown on Figures V-79 and V-80. These extremities were selected for the best contrast of temperature rise rates, $dT_{wg}/d\theta$, so as to increase the accuracy of the technique. An emissivity of 0.6 was assumed, conservatively high. A change to an emissivity of 0.3 signifies only a 50°F change in gas temperature. Averaging of the four thermocouples was used.

For each percent film coolant flow, a variation of gas temperature as a function of station resulted, typical of which is shown on Figure V-78. It was noted that at the lower film flow rates, the gas temperature in the chamber nearest the injector was higher than that at the throat. The reverse was found for higher film flow rates as in Figure V-78. Higher film flow rates exhibit increasing temperatures as the liquid film disappears with increasing travel distance.

The temperature of the throat station was used for the variation of temperature as a function of film flow rate in Figure V-81. Essentially, a straight line would represent the variation going through a point of 3500°F at zero percent film flow rate, which was attained for a radiation cooled thrust chamber tested at the start of the program.

(b) Water Cooled Test Data

The aluminum thrust chamber used for injector evaluation was evaluated for its heat transfer characteristics. Variations in gas temperature affect the heating rates nearly linearly while a variation in chamber pressure reflects itself by the 0.8 power, a predominant factor in the heat balance resistance.

A map of these variations for both nozzle and barrel is shown in Figure V-82.

Each test point was identified as to driving temperature and when plotted versus percent film flow rate, a significantly lower driving temperature resulted, 500°F less than for a less severely cooled boundary layer.

A comment is due concerning the two techniques for estimating the gas temperature at the wall from experimental test firings. One must remember that the preference is to operate with a wall essentially equal to the gas temperature, that is, adiabatic.

The rocket combustion gas is not an infinite source of heat and when heat is extracted by a large capacity cooling system, the total temperature of the gas is reduced by an amount equal to the extracted portion of the total available energy. A small unit of 100-lb thrust with a flat transverse temperature profile (with all the heat extracted uniformly from boundary layer to the nozzle axis centerline) will have its gas temperature reduced 120°F. For a larger engine, the effect is not as severe as indicated by the $\frac{\text{volume}}{\text{surface area}}$ ratio. Now, as the heat source is not infinite and the gas has a low thermal conductivity, a temperature profile can exist outside the formal (one dimensional) thermal boundary layer which would tend to destroy the previous assumption initially used.

Significant reductions in temperature of the order of 500°F are indicated which will alleviate material requirements for high temperature service. This has been done at both 80 and 135 psia in heat sink hardware and at 80 psia in water cooled hardware. These tests indicate that the injector does in fact provide the cool effective gas temperature at the wall. These tests also showed that 13 percent fuel film coolant provided approximately 3000°F effective gas temperature and 22 percent fuel film coolant provided an effective gas temperature below 2500°F. Data for the water cooled hardware has not been corrected for the cold wall operation; extrapolation of this data to a hot wall condition would superimpose the data on the stainless steel line. The accuracy of this method was examined and shown to be within 100°F for the conditions shown even if a 10°/sec error would have been made in defining the $dT_{wg}/d\theta$ slope from the test data. The slope was determined from an average of four thermocouples and its maximum deviation using the most divergent thermocouple was less than five degrees.

(5) Altitude Pulse Tests

Five pulse test series (Runs 87-91) were conducted in the altitude test chamber 2E-S at Bell Test Center. All tests were conducted on the Bell designed integrated thrust and flow measurement test stand pictured in Figure V-83. The hardware used for these tests consisted of injector TS-4 (split triplet) assembled to a stainless steel start chamber. The start chamber incorporated a flanged sea level nozzle with a 2.0 area ratio. Figure V-84 shows the thrust chamber assembly installed on the test stand. A Moog bipropellant valve (Model 50-245A torque-motor operated) completed the thrust chamber assembly.

The initial two tests were conducted at sea level conditions to provide a system checkout. The subsequent three runs were conducted at a pressure altitude of approximately 100,000 feet. Pulse widths of 45 and 65 milliseconds were fired with "off" times consistently in the order of 100 milliseconds. The first two vacuum pulse tests were limited to ten pulses each while the third test was extended to 25 pulses. A portion of a typical oscillograph is shown in Figure V-85.

Chamber pressure was monitored with a water-cooled Kistler pickup and a close-coupled Taber transducer. The Kistler was employed to determine the severity of dynamic vacuum ignition pressure spikes, while the Taber was used to follow steady-state pressure conditions. Pulse mode propellant flows were obtained by displacement measurements of small bellows tanks employed to supply propellants to the engine.

Thrust and flow measurements were compromised on these tests. The thrust capsule was damaged (apparently during test stand buildup) and internal leakage problems occurred in the propellant feed system. For these reasons, pulse mode specific impulse could not be determined.

Start and shutdown transients can only be estimated. The bipropellant valve current signal was not recorded. Only a control panel switch pip was monitored which lagged the feed pressure dropoff.



A few qualitative observations regarding the pulse traces are possible:

- (a) Random ignition overpressures did occur and were in the order of two to three times the rated chamber pressure of 80 psia. They disappeared after approximately 8 pulses.
- (b) The response time (signal to 90 percent of chamber pressure) was estimated to be approximately 17 milliseconds.
- (c) Thrust decay time to 10 percent P_c was approximately 7 milliseconds.
- (d) The injector provided stable combustion characteristics during all tests.

The indicated mixture ratio variation for the 45 millisecond pulses was between 1.4 to 2.0. The indicated mixture ratio variation for the 65 millisecond pulses was between 1.3 to 1.8. Rework to the gas bleed system on the bellows tanks is required to improve mixture ratio control and performance measurement. Modifications to the pulse control electronic circuitry are also required to allow greater pulse width variations.

(6) Vacuum Ignition Testing

During the second quarter this program was redirected to incorporate a more extensive investigation of the vacuum ignition characteristics of low thrust injectors. More specifically, emphasis was placed upon the design and development test of injector innovations which would reduce vacuum ignition lag and, hopefully, minimize the attendant ignition detonations or pressure spikes. Toward this end, a company sponsored project to investigate possible pressure spike suppression devices for low thrust injectors was integrated into this program.

As a basis for this portion of the hardware development and test program, a tentative working hypothesis regarding the vacuum ignition process of hypergolic propellants (of the N_2O_4 -hydrazine derivative category) was established. The hypothesis is based upon collective information from various investigations of vacuum ignition phenomena. This hypothesis, briefly stated, indicates that when the propellants (in this case N_2O_4 and MMH) are injected into a near vacuum pressure environment (less than the triple point pressure of the oxidizer or fuel), the vapor pressure of each liquid tends to shred the ligaments and larger

droplets of the injector spray pattern into much smaller droplets. It is assumed that the droplet disruption continues until a droplet size is reached where the surface tension can contain the internal vapor pressure forces. Simultaneously, the droplets evaporate and ultimately freeze in the process since no liquid-vapor equilibrium can exist at pressures below the triple point pressure. The time required for the droplets to freeze is exceedingly short (in the order of a few microseconds or less) and as a result, partially mixed solid (frozen) phase propellants accumulate in the thrust chamber. The statistical probability of ignition diminishes as the propellant temperature decreases. Ignition then, does not take place until the solid-gas phase equilibrium pressure buildup in the chamber exceeds the triple point pressure of the N_2O_4 (~ 2.7 psia) which is the critical (highest triple point pressure) propellant. At this point, propellant freezing ceases and liquid and gas phase reactions are promoted. Ignition may then occur at some random point which apparently sets off a detonation type of reaction in the accumulated propellant mixtures. This results in a characteristic pressure spike which precedes a normal chamber pressure rise.

It is necessary therefore, to adequately mix the propellants before freezing occurs or pressurize the combustion chamber above the most critical triple point pressure if the ignition delay and subsequent pressure spike are to be minimized. In context with this hypothesis, a highly confined vortex mixing tube device (preignitor) was selected as the primary design approach for investigation. Such a device is pictured schematically in Figure V-86.

The preignitor consists of a very narrow vortex tube in which fuel and oxidizer are injected tangentially. As the propellants enter this tube, and before they start reacting, a liquid backpressure of more than 10 psia is built up. This prevents initial freezing in the tube. The swirling mixture in the tube undergoes partial reaction; however, the tube diameter is in the order of the quenching distance and the stay time is such that full combustion reaction does not occur in the tube. Therefore, the products of this vortex tube are largely a mixture of low temperature gases in the order of several hundred °F. The gas generated from the vortex tube aids in pressurizing the main combustion chamber rapidly to a level such that freezing of the propellants from the main injector will be minimized.

The prereacted gas mixture also contains a large number of molecules at higher energy levels which act as ignition kernels to raise the ignition probability, thus reducing the statistical scatter of ignition delays. Since the maximum temperature of the device is low, there will be no heating problems within the injector face due to its operation. In addition, this device is readily incorporated in the existing Model 8374 injector designs.

Three stainless steel preignitors were designed and fabricated. The initial unit was made to conduct water flow tests. Photographs of a typical water flow test are shown in Figures V-87 and V-88. The photographs show the preignitor flow model in two planes approximately 90° apart. From these pictures, the conical spray pattern emanating from the vortex tube may be observed.

Following the water flow tests, the preignitor was fire tested into the atmosphere. A total of four tests was made using N₂O₄ and MMH propellants. Figure V-89 is a photograph of the preignitor firing. The purpose of these tests was only to determine whether propellant reaction would occur upstream of the vortex tube and thus cause an explosion in one of the feed tubes. This did not occur. Temperature measurements were monitored on each of the hypodermic feed tubes and on the vortex tube exterior as illustrated in the photograph of Figure V-89.

A summary of the temperature data from these tests is presented below:

Preignitor Element
S/N-1
Atmospheric Fire Test Data

Run No.	Run Duration in Sec.	Temperatures in °F				
		Ambient Prop. Temp.	Preignitor Temp			
		<u>Oxid.</u>	<u>Fuel</u>	<u>Oxid. Line</u>	<u>Fuel Line</u>	<u>Vortex Tube</u>
2004	2	62	57	57	89	71
2005	10	62	62	62	167	80
2006	5	60	62	71	62	132
2007	5	60	62	62	62	158

A full scale (100-lb thrust) injector S/N CS-1 with an integral preignitor element was designed and fabricated and subjected to vacuum ignition starts. The injector, pictured

in Figure V-90, consisted of a six-element triplet design ($F_{vac} = 100$ lb) with one of the triplets removed and a preignitor substituted in its place. A Moog Model 50 bipropellant valve completed the test assembly.

The test injector was assembled to a stainless steel start chamber as shown in Figure V-91. The chamber had provisions for a flanged attachment to a vacuum tank. This tank, equipped with a blow-off cover at the opposite end, was evacuated by means of a vacuum pump to pressure altitudes in excess of 150,000 feet. With this test setup, a series of vacuum ignition tests was conducted. See Figure V-92.

The results from the preignitor injector tests are tabulated in Table V-10. As can be seen from this tabulation, the preignitor design used with this injector had no appreciable effect in either suppressing the ignition spike or reducing vacuum ignition delay. The preignitor did introduce low frequency instability into the thrust chamber operation. Comparative pulse traces (oscillograph and oscilloscope photographs) are reproduced in Figures V-93 and V-94. In Figure V-93, sea level and vacuum pulses are shown which represent injector operation without the preignitor. The pulses in Figure V-94 illustrate the results of operating the injector with the preignitor. The chamber pressure oscillations are in the order of 300-400 cps with a double amplitude approaching rated (80 psia) chamber pressure.

Following the above test program, the injector assembly (including the Moog bipropellant valve) was set up on the water flow stand. A series of water flow pulse tests were made (flowing at rated conditions to the atmosphere) and recorded on high speed (approximately 8000 frames/second) motion picture film to determine if flow emitting from the preignitor was leading or lagging that of the main injector. The results of these tests indicated that the preignitor was lagging the main injector by about one millisecond.

The second preignitor element with a slightly enlarged vortex mixing tube bore was tested to evaluate the effect of reduced back pressure (in the mixing tube) on combustion stability. A drawing of the preignitor assembly is shown in Figure V-95. Since the total propellant weight flow through this unit was approximately 0.04 pound/second, a small Moog bipropellant valve (Model 50-259) was substituted for the larger Moog valve to complete the ignitor assembly.

TABLE V-10
SPIKE EVALUATION AND SUPPRESSION TESTS
Injector S/N CS-1

Without Suppression Device					Prop. Temp +60°F		With Suppression Device				Prop. Temp +60°F		
Pulse	Altitude Feet (Microns)	Ignition Delay Msec	Start Delay 100% P _c	Recorded Peak Press psia	Estimated Peak Press psia	Pulse	Altitude Feet (Microns)	Ignition Delay	Start Delay	Recorded Peak Press psia	Estimated Peak Press psia		
2072	Sea Level	11.0	23.5	No Spike									
2073	240,000 (25)	16.5	24	156	123	2067	230,000 (50)	16.5	-	133	100		
2074	160,000 (700)	15.5	23	178	135	2068	170,000 (500)	17	-	190	122		
2075	160,000 (700)	16.0	23.5	195	155	2069	160,000 (700)	17.5	-	200	128		
2076	180,000 (450)	15.5	23.5	370	200	2070	160,000 (700)	17.5	-	167	128		
2077	165,000 (600)	16.5	23	190	140	2071	160,000 (700)	17.0	-	100	67		
Prop. Temp +32°F													
						2050	258,000 (20)	16	-	330	167		
						2051	215,000 (850)	19	-	278	178		
						2052	155,000 (900)	16.5	-	268	190		
						2053	155,000 (900)	17.5	-	220	155		
						2054	155,000 (900)	16	-	334	190		

A series of vacuum ignition tests was conducted with this preignitor assembly. The low frequency instability experienced with the previous unit persisted with this preignitor. Testing was suspended and no further single element preignitors were tested.

An alternate method of pressurizing the combustion chamber to a pressure level above the critical triple point pressure is the employment of a very small combustion chamber immediately upstream of the main chamber. Termed a "precombustor", this device acts as a gas generator to supply pressurization gas to the (main) combustion chamber. The precombustor may be differentiated from the preignitor element by the degree of reaction or combustion of the propellants which takes place. In the preignitor there is a relatively low order reaction resulting in low temperature partially reacted gases. The precombustor, on the other hand, supports a propellant reaction approaching full combustion with correspondingly high gas temperatures.

A precombustor assembly (see Figure V-96) was designed and fabricated for vacuum ignition testing. The design of the precombustor was based upon information furnished by NASA-MSC. The small Moog bipropellant valve was again employed.

The precombustor assembly was bolted to a stainless steel start chamber. The test stand setup was the same as that used with the preignitor.

The initial testing with this unit was done with feed pressures of 180 and 50 psi on the oxidizer and fuel, respectively. Flows were not measured. Feed pressures for the rated design conditions were established through water flow tests.

Following an initial sea level checkout firing, five vacuum starts were conducted. The oscillograph trace presented in Figure V-97 is typical of the test results. It can be seen that a pressure rise occurs (in the main chamber) some 20 milliseconds after the valve signal. This is indicative of some low order propellant reaction and vapor pressure buildup. At the end of 47 milliseconds, an ignition pressure spike is generated followed by a steady-state combustion pressure of 7.8 psia. Propellant inlet temperatures for these tests were approximately 40°F.

A second series of vacuum starts was initiated with the precombustor assembly. This time, the feed pressures for oxidizer and fuel were equalized at 180 psi creating a highly fuel rich condition.

These tests were conducted to simulate a more reasonable design feed pressure condition which would be encountered with the Model 8374 engine operation. The drastic change (increase) in fuel feed pressure produced strong low frequency stability (300-600 cps) operation. The test series was discontinued.

(7) Injector Test Results - Columbium Chamber

Test injectors TF-2, TU-3, and TS-4 were fired in the radiation-cooled columbium alloy (CU 121) thrust chamber S/N 1. The purpose of these tests was to:

- (a) Determine injector performance using a "hot wall" test chamber.
- (b) Establish resultant steady-state external surface temperatures under free radiating conditions.
- (c) Obtain more complete information for calculating gas driving temperatures adjacent to the combustion chamber wall.

Test injector TF-2 was first fired without (fuel) film coolant flow for a total of three tests. The individual film coolant splash plates utilized on this injector were distorted during the course of previous firings. Before testing with this injector was resumed, the damaged film cooling elements were removed and new units installed. The replacement film cooling elements were modified to permit direct impingement of the free flowing fuel streams on the chamber wall. The nominal flow rate of the new design was unchanged. In summary, the splash plate element design was abandoned because of the heating loads imposed upon this configuration.

The modified injector, designated as TF-2A (Figure V-98), was again assembled to the columbium alloy thrust chamber and fire tested. A total of five tests was conducted at film cooling flow rates of approximately 12 and 20 percent. The average nozzle temperature and injector performance (c^*) values for all TF-2/TF-2A "hot wall" chamber tests are presented in Figure V-99 as a function of the film coolant flow.

For these hot wall columbium chamber tests, all the performance data are corrected for thermal expansion of the throat and location of the chamber pressure pickup. (Chamber pressure was sensed at the injector face and corrected for total pressure at the nozzle.) These corrections consist of a throat area increase of 2.0 percent at 2500°F and a chamber pressure and characteristic velocity decrease of 1.2 percent for a contraction ratio of 4:1, as shown in Figure V-100.

Injector TU-3 was next assembled to the columbium test chamber. A series of six firings was conducted at various mixture ratio and chamber pressure conditions. Injector performance was quite low with combustion efficiencies in the order of 85.89 percent of shifting theoretical.

The injector was removed from the test stand and subjected to another water flow test. It was observed that the direction of the impingement fan pattern was not axial. When the impinging fuel orifices were water flowed separately, it could be seen that the resultant fan momentum was unbalanced (~ 10 to 15 degrees) in the direction of the chamber wall. This indicated that the mass flow through the inner fuel orifices was greater than that flowing through the corresponding outer fuel orifice of each triplet element. To approach a more axial resultant momentum condition of the impinging fuel streams, an orifice was added immediately downstream of the valve/injector interface in the large hypodermic tube feeding the inner ring of fuel orifices. In this manner, the flow impedance of the feed circuit to the inner fuel orifices was increased while the outer fuel feed system impedance remained unchanged.

The injector thus modified (designated TU-3A) exhibited a somewhat more axial spray pattern although the unbalanced momentum characteristic was still apparent. Fire tested in this condition, the TU-3A injector showed a performance improvement of approximately 1-1/2 percent.

The injector was again disassembled and the flow impedance further increased in the feed circuit to the inner orifices. The result was a more nearly axial resultant momentum condition although a few degrees of imbalance still existed.

The injector was redesignated TU-3B and again fire tested. Five runs were made with no discernible change in performance over the previous TU-3A configuration. The combustion efficiency of the various TU-3 injector configurations is plotted in Figure V-101 as a function of mixture ratio. It is important to note that the TU-3 type injector is capable of reasonably high performance (combustion efficiency) with the increased fuel injection velocities of fuel rich mixture ratio operation.

At a 1.23 mixture ratio, the combustion efficiency was 91.7 percent of the shifting theoretical reference. In terms of combustion efficiency this is only 1.8 percent below the required value of 93.5 percent necessary to reach a 290-second specific impulse at a 1.6 mixture ratio. It is significant that this increase in combustion efficiency did not appreciably affect the low wall temperatures ($< 2200^{\circ}\text{F}$) produced by the TU-3 series injector. This can be seen graphically in Figure V-102 by comparing the performance (c^*) and stabilized temperature spreads for the TU-3 injector firings.

Injector TS-4 was next fired in the columbium test chamber. This injector again exhibited the highest performance attaining a combustion efficiency of 96.6 percent of the shifting theoretical c^* reference at a 1.6 mixture ratio. The nozzle temperatures were correspondingly high reaching approximately 2800°F in fifteen seconds run time. Thermal equilibrium was not reached prior to shutdown. No further long duration runs were attempted so as to avoid damaging the columbium chamber. Based upon the rate of rise for various skin temperatures during the fifteen-second run, a stabilized wall temperature of approximately 3000°F was extrapolated.

Injector performance as a function of mixture ratio for all "hot wall" firings is shown in Figure V-103. A tabulation of the hot wall data is given in Table V-11.

Thrust chamber skin temperatures were monitored during all firings for each injector series. Temperatures were measured in the chamber barrel section and at the nozzle throat using platinum/platinum-rhodium thermocouples. During all steady-state tests, a conventional heating pattern was observed with the highest temperatures in the vicinity of the nozzle convergence and throat. The maximum and minimum temperatures recorded at

TABLE V-11
MODEL 8374 INJECTOR SEA LEV
COLUMBIUM ALLOY THRUST C

Run No.	Inj. S/N	Dur. Sec.	Data P _t Sec.	% FC ¹	O/F	P _c psia
2119	TF-2	10.5	10.0	0	1.55	78.4
2120	TF-2	14.5	14.0	0	1.55	77.9
2121	TF-2	30.5	30.0	0	1.60	80.9
2128	TF-2A	10.5	10.0	12.0	1.55	78.3
2129	TF-2A	17.5	17.5	11.8	1.56	78.7
2130	TF-2A	29.5	29.0	11.5	1.54	78.1
2131	TF-2A	30.5	30.0	19.3	1.55	75.7
2132	TF-2A	30.5	30.0	19.6	1.38	81.0
2122	TU-3	10.5	10.0	-	1.55	78.1
2123	TU-3	45.8	45.3	-	1.54	77.2
2124	TU-3	31.5	31.0	-	1.32	79.9
2125	TU-3	10.5	10.0	-	1.60	96.2
2126	TU-3	10.5	10.0	-	1.64	123.0
2127	TU-3	31.5	31.0	-	1.41	128.1
2133	TU-3A	10.5	10.0	-	1.50	77.2
2134	TU-3A	30.5	30.0	-	1.54	77.7
2135	TU-3A	30.5	30.0	-	1.34	79.9
2136	TU-3B	10.5	10.0	-	1.57	78.1
2137	TU-3B	30.5	30.0	-	1.56	77.7
2138	TU-3B	30.5	30.0	-	1.32	81.5
2139	TU-3B	30.5	30.0	-	1.73	75.5
2140	TU-3B	10.5	10.0	-	1.23	81.5
2157	TS-4	6.0	5.0	-	1.52	74.3
2158	TS-4	10.5	10.0	-	1.56	80.3
2159	TS-4	15.5	15.0	-	1.58	79.9
2160	TS-4	5.5	5.0	-	1.79	76.3
2161	TS-4	5.5	5.0	-	1.36	77.0
2162	TS-4	5.5	5.0	-	1.25	76.3

1. Percent of total fuel flow diverted to film cooling.

2. c* corrected for thermal expansion of the nozzle throat and chamber pressure pickup location

V-124-1

EL TEST RESULTS
CHAMBER S/N 1

c^*_{cor} ² ft/sec	η_c %	Calc. Driving Gas Temp. (°F) (Ave.)		Max. Chamber Temp. (°F)	Max. Thrust Temp. (°F)
		Chamber	Throat		
5325	93.5				
5330	93.5				
5315	93.4	3170	3380	2800	2950
5290	92.5				
5330	93.5				
5310	93.2	2800	3185	2400	2830
5100	89.5			1730	2695
5225	92.3			2135	2770
4905	86.0				
4880	85.7	1470	1835	1545	1925
5020	89.1	1590	1835	1675	1925
4900	85.8				
4790	83.8				
5010	88.2	1800	2105	1945	2120
4990	87.5				
4990	87.4			1775	1910
5095	90.3			1835	1910
4970	87.0				
4970	87.1	1645	1790	1690	1885
5125	90.9	1735	1820	1770	1920
4810	84.3			1575	2080
5125	91.7				
5470	96.2				
5510	96.5				
5520	96.6				
5430	95.4				
5370	95.0				
5385	94.4				

the throat station for each steady-state duration firing are illustrated in Figure V-102. For comparison, these temperature ranges are related to performance. Figure V-104 is a schematic diagram of the thermocouple locations for the tests described in the previous paragraphs.

(8) Hot Wall Chamber Analysis-Driving Gas Temperatures

The data from the above tests were used to calculate the driving gas temperatures adjacent to the thrust chamber wall.

Knowing the geometries of the chamber, material properties, and the temperature time history of the thrust chamber wall it is possible to infer the effective gas temperature and film coefficients in the chamber and in the nozzle. At any instant, the heat added to an element of the chamber is equated to the time rate of temperature rise of a thermal mass plus the radiant heat lost by the outer wall to its environment.

The following equation is examined at two instants during the firing, once shortly after the heat pulse has penetrated the chamber wall - $\theta = 4$ seconds and again near the shutdown time.

$$h_g (T_g - T_{wg}) - \sigma \epsilon \frac{A_2}{A_1} T_{wg}^4 = \text{Thermal Mass} \frac{dT_{wg}}{d\theta}$$

The simultaneous solution of these equations then yields the desired T_g . An emissivity of 0.6 was assumed as a rather conservatively high value.

Typical transients plots from digital test data appear in Figures V-105, V-106, and V-107. These characteristic thermocouple responses are used to predict the driving potentials.

The derived data are included in Table V-11 for an array of injector configurations. It should be noted the calculated driving gas temperatures indicated in this table are average gas temperatures, not maximum temperatures.

b. Injector TF-5

(1) Injector Description - TF-5

The fifth injector design established for this program was a variation of test injector TS-4. Designated as TF-5, this injector design was an attempt to retain the high performance level (in excess of 94% of shifting theoretical c^*) of injector TS-4 and at the same time, lower the heat rejection rates to a point compatible with adiabatic wall operation.

The intended purpose of the split triplet design was to effect a mixture ratio gradation across each element by splitting the central oxidizer flow into two streams. In this way, the mass flow through the two central oxidizer orifices could be varied so that a different mixture ratio setting was established with the corresponding inner and outer impinging fuel streams. The rated design mixture ratio condition for the TS-4 injector was 1.80 and 1.40 respectively for the inner and outer half of each split triplet element. The incoming or average mixture ratio would be 1.60. The outer half of each triplet, adjacent to the thrust chamber wall, would provide a shroud of combustion gases burning at a 1.40 mixture ratio around the hotter burning core.

A close examination of the TS-4 split triplet water spray pattern revealed a strong secondary impingement of the initial fans from each half of the triplet. It was theorized, in effect, that the intended mixture ratio distribution or gradation between inner and outer portions of the elements was destroyed during the secondary mixing phenomena.

As indicated previously, the TF-5 injector design is another attempt to physically promote a mixture ratio gradient across each element by splitting the oxidizer flow and forming a fuel rich mixture between the chamber wall and core gases. To eliminate the secondary fan impingement and, hopefully, preserve the initial mixture ratio distributions, three significant changes were incorporated in the new design. First, the mixture ratio of the outer half of the split triplet was designed for a 1.10 condition as compared to the 1.40 ratio of the previous injector.



Secondly, film cooling orifices were provided for in the design so that the local mixture ratio conditions at the wall could be further reduced if necessary. The film cooling orifices were sized for 11 percent of the total fuel flow at rated conditions. On the test injector, the feed system to the film cooling orifices is separate from the main injector feed network. In this manner, the film cooling can be controlled separately and the heat rejection qualities of the injector can be evaluated either with or without the additional cooling capacity of the film. And finally, the oxidizer orifices in each element were angled such that the resultant fan momentum of each half of the split triplet is essentially axial. As a result, heavy secondary fan impingement will be avoided. Thus, effectively, the injection element is split in two parallel doublets. This injector is shown in Figures V-108 and V-109.

An additional feature of injector TF-5 is the incorporation of a revised spike suppression device. This device is a variation of the original preignitor design which attempts to alleviate much of the back pressure believed responsible for the unstable operation of the first test unit. In addition, a physical propellant lead to the preignitor was incorporated by employing direct feed capillary tubes to the injection orifices. The propellants destined for the main or primary injector are first routed through manifolds which in turn feed individual capillary tubes leading to the respective oxidizer and fuel orifices. This feed network is also shown in the schematic of Figure V-109. The designed propellant distribution for the TF-5 injector is as follows:

TF-5 Injector Propellant Distribution

<u>Item</u>	<u>Fuel lb/sec</u>	<u>Oxidizer lb/sec</u>	<u>O/F</u>	<u>% of Total Flow</u>
Film Cooling	0.0140	-	-	4.0 (10.7% of Fuel Flow)
Preignitor	0.0144	0.0230	1.60	11.0
Split Triplet				
Outer Half	0.0687	0.0756	1.10	42.5
Inner Half	<u>0.0339</u>	<u>0.1104</u>	3.26	<u>42.5</u>
Total	0.1310	0.2090		100.0%



(2) Injector Test Results

Columbium chamber injector TF-5 was fire tested in the radiation cooled columbium chamber S/N 1.

The injector was initially tested in the radiation cooled columbium chamber with the preignitor in operation and without fuel film coolant flow. A total of nine tests were completed in the series with a maximum duration of 15 seconds and with variations in mixture ratio and chamber pressure. No instability was noted during this series. A tabulation of the data are given in Table V-12. Although the chamber skin temperatures did not reach equilibrium conditions, the average stabilized temperature is projected to be between 2800 and 2900°F with a characteristic velocity of approximately 5330 ft/sec (93.3% combustion efficiency).

Just prior to shutdown of the last test, which was conducted at a chamber pressure of approximately 91 psia, slight sparking was noted at the nozzle exit. Post run inspection revealed erosion in the convergent nozzle to throat section at approximately the 4 o'clock position. A water flow was performed on the injector revealing misimpingement of one triplet element which was nearly in line with the eroded section of the nozzle. Backflush and reflowing of the injector revealed additional misimpingement problems attributed to contamination of the injector fuel system. The injector was steam cleaned and reflowed revealing desirable impingement characteristics.

(3) Injector Test Results - Stainless Steel Chamber

Although the damage to the columbium chamber was not severe, further testing in the columbium chamber could result in questionable performance data. For that reason the test series was completed with the stainless steel chamber and run durations were generally limited to 5 seconds. This test series consisted of 13 tests; four tests with the preignitor plugged and without film coolant flow; five tests with the preignitor plugged and with approximately 15% film coolant flow; two tests with the preignitor in operation and with approximately 13% film coolant flow; and two tests with the preignitor in operation and without film coolant flow. These data are also tabulated in Table V-12. The data for this injector is also given in Figure V-110.

TABLE V-12
INJECTOR TF-5 DATA SUMMARY

Columbium Chamber Tests

Run No.	Duration (sec)	Condition	O/F	c* corr (ft/sec)	P _c (psia)
2166	5	W/P-I, 0% F.C.	1.50	5330	75.8
2167	10	W/P-I, 0% F.C.	1.68	5315	81.1
2168	3	W/P-I, 0% F.C.	1.67	5285	81.2
2169	15	W/P-I, 0% F.C.	1.62	5325	80.9
2170	15	W/P-I, 0% F.C.	1.32	5305	80.9
2171	15	W/P-I, 0% F.C.	1.47	5310	80.5
2172	15	W/P-i, 0% F.C.	1.80	5235	78.8
2173	15	W/P-I, 0% F.C.	1.65	5205	66.0
2174	15	W/P-I, 0% F.C.	1.59	5310	91.0

Stainless Steel Chamber Tests

2175	3	W/O P.I. 0% F.C.	1.57	5200	77.7
2176	5	W/O P.I. 0% F.C.	1.60	5230	81.2
2177	5	W/O P.I. 0% F.C.	1.33	5195	81.0
2178	5	W/O P.I. 0% F.C.	1.82	5190	80.7
2179	5	W/O P.I. 14.6%	1.60	5125	80.1
2180	1	W/O P.I. 14.6%	1.60	5100	79.3
2181	5	W/O P.I. 14.7%	1.35	5185	81.8
2182	5	W/O P.I. 14.7%	1.83	5080	80.0
2183	10	W/O P.I. 14.7%	1.60	5135	79.0
2184	5	W/P.I. 13%	1.61	5175	81.2
2185	1	W/P.I. -	1.60	5210	81.0
2186	5	W/P.I. 0%	1.63	5280	83.2
2187	1	W/P.I. 0%	1.62	5360	83.5

The data obtained in the columbium chamber and in the stainless steel chamber (with the preignitor in operation and without film coolant) agree within 1%. The performance decreased approximately 1.9% when the injector was tested with the preignitor plugged and without film coolant. Chamber surface temperature effects could not be accurately determined with the stainless steel chamber.

(4) Spike Suppression Tests

Injector TF-5 was subjected to a series of 100 millisecond pulses at a simulated pressure altitude of 200,000 feet (minimum) both with and without the preignitor in operation. These tests were conducted with the stainless steel chamber.

As previously discussed, the preignitor incorporated in this injector is a variation of the original preignitor design and attempts to reduce the back pressure believed responsible for the unstable operation of the initial preignitor. Also, a physical propellant lead to the preignitor is assured by use of direct feed capillary tubes to the preignitor injection orifices and by routing the propellants destined for the primary injector first through manifolds which in turn feed the individual capillary tubes leading to the oxidizer and fuel orifices.

A total of fifteen 100 millisecond pulses were made during this Bell sponsored test series. During these tests the propellants were conditioned to $+32^{\circ}\text{F}$. The initial tests were conducted with the preignitor in operation and consisted of one sea level pulse, four altitude pulses with the hardware at ambient temperature, and four altitude pulses with the injector at approximately 0°F and the chamber at approximately -65°F . The remaining tests were conducted with the preignitor plugged and consisted of one sea level pulse and four altitude pulses with ambient hardware, and one altitude pulse with the injector at approximately 0°F and the chamber at approximately -85°F . The start transient of this last pulse is not representative of the conditions due to leakage at an oxidizer capillary tube feeding one of the primary injector orifices. No overpressure or spike was evident during this pulse. Although no definite cause was found to explain the tube leakage, some of the capillary tubes showed external scoring and damage due to assembly techniques, resulting in probable reduction of the cross section of the thin wall tubes (0.010 inch wall thickness).

The results of this test series are shown in Table V-13 and reproductions of the oscillograph traces are given in Figures V-111 through V-125.

Comparison of the tests conducted with and without the preignitor in operation and with ambient hardware reveals that the preignitor successfully suppresses the spike by approximately 50%. Unfortunately, a comparison cannot be made with and without the preignitor with the hardware at low temperatures due to the lack of test results with the preignitor plugged. The first three pulses conducted with the preignitor and with low temperature hardware reveal spikes less than 100% chamber pressure while the fourth and last pulse reveals a spike with an estimated peak of 290 psia. It is possible the variation in these start transients may be attributed to partial freezing of the oxidizer due to low injector temperatures. There is no evidence of low frequency instability during any of the pulse tests as originally encountered in the initial design.

During these tests chamber pressure was recorded using a water cooled, flush mounted Kistler transducer and a close coupled Alinco transducer. Oscilloscope photographs were also taken of the start transients and were used to determine the values of the indicated and estimated peak pressures. Reproductions of these photographs are also included in Figures V-111 through V-125.

Following this series this injector TF-5 was refurbished by replacing all the capillary tubes with heavier wall tubing. Checkout tests were made at a simulated altitude of approximately 200,000 feet prior to shipment of this injector to NASA-MSC, Houston for evaluation at that facility. The results of these checkout tests showed overpressures of the same magnitude as described above. The evaluation of this injector at NASA-Houston will be performed after the reporting period of this contract.

c. Injector TU-6

(1) Injector Description - TU-6

This sixth injector is based upon an iteration of the unbalanced triplet design of test injector TU-3. As described earlier, the unbalanced triplet design yielded exceptionally

TABLE

MODEL

VACUUM START

Injector TF-5

Run No.	Steady State ⁽¹⁾ P _c (psia)	Indicated Peak Press ⁽²⁾ psia	Estimated Peak Press ⁽²⁾ psia	Start Response Milliseconds			Shutdown Transient Milliseconds D	Mic
				A	B	C		
2195	74.1	-	-	13.8	21.4	25.0	5.7	S/
2196	76.7	211	166	20.0	25.2	30.2	5.6	22
2197	77.8	152	130	19.9	23.9	29.7	5.7	53
2198	78.7	225	194	19.1	24.6	30.0	5.7	40
2199	78.7	132	124	21.1	23.4	29.6	5.8	53
2200	78.7	45	39	17.6	20.9	29.8	5.7	23
2201	79.2	59	51	-	19.6	34.2	5.6	53
2202	79.2	56	51	18.3	20.6	30.7	5.7	50
2203	79.7	361	290	17.9	24.5	29.4	6.0	50
2205	75.6	98	98	16.2	19.6	23.0	4.9	S/
2206	83.9	496	456	22.0	27.6	31.9	5.2	11
2207	81.3	434	338	20.6	24.9	29.1	5.5	50
2208	80.8	372	282	19.7	24.5	29.3	5.3	50
2209	80.8	256	223	20.1	24.2	28.4	5.4	50
2210	68.4	-	-	-	19.3	35.8	4.9	30

(1) Values obtained from Alinco trace on speed oscillograph trace.

(2) Values obtained from oscilloscope photo of Kistler trace.

(3) Injector temperature approximately 600°F

Valve Signal

Pressur
Peak

Chamber Pressure

A →

B →

(100% P

11-132-1

V-13

8374

TEST RESULTS

N_2O_4 -MMH

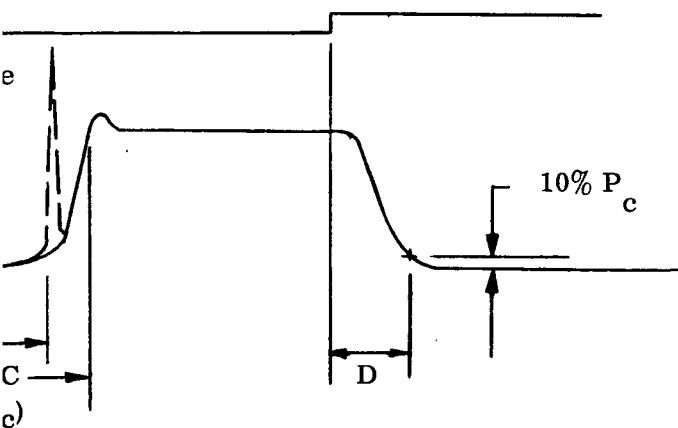
Altitude		Run Duration Milliseconds	Prop Temp °F	Thrust Chamb. Temp. °F	Remarks
ons	Feet				
/L	S/L	100	32	Ambient	Injector TF-5 with Preignitor
2	246,000	100	32	Ambient	Injector TF-5 with Preignitor
5	227,000	100	32	Ambient	Injector TF-5 with Preignitor
0	223,000	100	32	Ambient	Injector TF-5 with Preignitor
5	227,000	100	32	Ambient	Injector TF-5 with Preignitor
5	243,000	100	32	-65(3)	Injector TF-5 with Preignitor
5	227,000	100	32	-65(3)	Injector TF-5 with Preignitor
0	229,000	100	32	-65(3)	Injector TF-5 with Preignitor
0	229,000	100	32	-65(3)	Injector TF-5 with Preignitor
/L	S/L	100	32	Ambient	Injector TF-5, Preignitor Plugged
3	250,000	100	32	Ambient	Injector TF-5, Preignitor Plugged
0	229,000	100	32	Ambient	Injector TF-5, Preignitor Plugged
0	229,000	100	32	Ambient	Injector TF-5, Preignitor Plugged
0	229,000	100	32	Ambient	Injector TF-5, Preignitor Plugged
0	238,000	100	32	-85(3)	Injector TF-5, Preignitor Plugged

high

otographs

0° to -5° F.

{ Indicated peak represents highest overshoot value of Kistler. Estimated peak represents mean of three highest amplitudes of Kistler.



low wall temperatures over a wide range of mixture ratio operation. Further, the combustion efficiency improved considerably (refer to Figure V-101) at the fuel rich operating conditions reaching 91.7% at a 1.23 mixture ratio. For these reasons, a modification of this design was desirable.

The tendency of the unbalanced fuel-on-oxidizer triplet to produce higher combustion efficiencies at more fuel rich mixture ratios is attributed to the increased fuel injection velocities imposed at these conditions. The TU-3B configuration (see the description of this injector in the previous section) was selected to serve as the basis for the sixth injector design. The TU-3B injector exhibited the highest performance (approximately 91.7% at O/F = 1.23) while maintaining the low wall temperatures characteristic of all TU-3 injector configurations.

The impedance of the inner fuel orifice feed system on injector TU-3B had been increased to obtain a more nearly axial resultant momentum condition on the propellant spray pattern. Because of this modification, a water flow distribution test was conducted to determine the fuel flow distribution between the inner and outer ring of fuel orifices. The original injector (TU-3) configuration possessed a design fuel flow distribution ratio of 1.3 $\frac{\dot{W}_{f_{outer}}}{\dot{W}_{f_{inner}}}$ between the outer and inner fuel orifices of each individual element. With the modified inner fuel orifice feed system, the water flow tests indicated a distribution ratio of 1.57 now existed.

To achieve an engine specific impulse of 290 seconds, the injector must achieve a combustion efficiency of 93.5% (shifting) at a 1.6 mixture ratio with a 40 area ratio nozzle. For this reason, the combustion efficiency curve developed for injector TU-3B was extrapolated (on a straight line basis) to a 93.5% value, some 1.8 percent beyond the last test point. This extrapolated combustion efficiency value corresponded to a mixture ratio of approximately 1.1. Using this information, a new total weight flow was calculated. Injection velocities (fuel and oxidizer) were determined based upon a 1.1 mixture ratio with the fuel distribution ratio set at 1.57 as determined by water flow test.

The injection velocities thus determined were translated into orifice diameters at a 1.6 mixture ratio condition while maintaining the 1.57 fuel flow distribution. The resulting injector design is pictured in Figure V-126. This injector has been designated as TU-6. Design details of this injector and the previous injectors are given in Appendix C.

(2) Injector Test Results - Columbium Chamber

Injector TU-6 was fire tested in the second radiation cooled columbium chamber S/N 2.

The test series on injector TU-6 consisted of 27 tests and was conducted to determine injector performance and chamber surface temperatures. Of these 27 tests, 22 were conducted in the modified configuration with the injector identified as TU-6A. This modification consisted of orificing the central fuel tube feeding the inner ring of fuel orifices. This resulted in a more axial spray pattern by increasing the flow impedance to the inner fuel orifices.

The performance of this injector is shown in Figure V-127 with an average c^* of 5430 ft/sec at a mixture ratio of 1.6. The combustion efficiency is 94.5% of the shifting theoretical reference and equivalent to a $I_{sp} = 293$ seconds at a 40:1 area ratio. The average c^* at a mixture ratio of 1.4 is 5460 ft/sec or 96.5% combustion efficiency and an $I_{sp} = 296$ seconds. Each data point is plotted as an ellipse to indicate maximum data error, allowing for a mixture ratio error of $\pm 1.03\%$ and a c^* error of 1.28%. The maximum temperature obtained during a stabilized 30 second duration test was 2340°F at the throat and 2360°F at the convergent nozzle section. The average throat temperature was 2220°F with a minimum temperature of 2050°F . The comparison of the performance-temperature characteristics of this injector with the previous injectors is shown in Figure V-128. The temperature stability characteristics of the throat station is shown in Figure V-129, and the temperature profile of the entire chamber at four different stations along the chamber and at six positions around the circumference is shown in Figure V-130.

All performance data were corrected for throat expansion and location of the chamber pressure pickup. (Chamber pressure was sensed at the injector face and corrected for total pressure at the nozzle).

A tabulation of the test data for injector TU-6A is shown in Table V-14.

The results of testing the TU-6 injector in the "as fabricated" configuration resulted in temperatures similar to those obtained when modified with performance approximately 1% lower.

This injector design definitely meets the original objective to obtain performance in excess of 5350 ft/sec characteristic velocity (290 second specific impulse at an area ratio of 40:1) with low wall temperatures. The average calculated driving gas temperatures for this injector is approximately 2700° F, derived from the average temperature measurements of 2300° F in the radiation cooled condition. Further definition of this driving gas temperature will be made later in this section during demonstration testing of the insulated chamber.

3. Adiabatic (Insulated) Wall Demonstration

a. Injector TU-3B - Columbium Chamber

To initially demonstrate the feasibility of running a refractory metal thrust chamber at essentially adiabatic wall conditions, injector TU-3B was installed in the columbium test chamber S/N 1. This chamber was then installed into an insulated shell assembly. The shell assembly was filled with alumina bubbles to provide an insulated barrier between the thrust chamber wall and the external surface of the assembly. The thrust chamber/shell assembly is shown schematically in Figure V-131. The shell assembly was fabricated in segments to permit satisfactory installation of thermocouple leads to the barrel and nozzle sections of the thrust chamber. A photograph of this assembly installed on the test stand is shown in Figure V-132.

During the test firings, temperatures were monitored on the chamber, nozzle, injector plate, shell assembly (internal wall) and the insulation at the external wall. The thermocouple locations are shown in Figure V-131. To supplement the thermocouple temperature measurements on the thrust chamber, a viewing port (see Figure V-131) was provided through the shell assembly to permit the use of an optical pyrometer.

TABLE V-14

INJECTOR TU-6A PERFORMANCE DATA
COLUMBIUM CHAMBER - RADIATION COOLED

Data corrected for throat expansion and contraction ratio (P_c pickup location)

Run No. (Cell B-2)	Run Duration (sec)	Data P_T (sec)	c^* corr (ft/sec)	P_c (psia)	O/F	Nozzle Min.	Temperature Ave.	(°F) Max.
2211	5	4.0	5608	80.1	1.45	-	-	-
2212	10	4.0	5420	80.9	1.56	-	-	-
2213	15	8.0	5444	80.6	1.55	-	-	-
		4.0	5465	79.7	1.63	-	-	-
		12.0	5428	78.2	1.62	-	-	-
2214	15	15.0	-	-	-	1450	1802	1919
		4.0	5540	80.2	1.47	-	-	-
		12.0	5483	79.4	1.48	-	-	-
2215	15	15.0	-	-	-	1473	1791	1957
		4.0	5492	79.7	1.25	-	-	-
		12.0	5476	79.2	1.26	-	-	-
2216	15	15.0	-	-	-	1462	1763	1909
		4.0	5438	79.4	1.19	-	-	-
		12.0	5440	78.8	1.18	-	-	-
2217	30	15.0	-	-	-	1442	1736	1893
		4.0	5435	80.7	1.54	-	-	-
		12.0	5422	80.0	1.54	-	-	-
2223	15	15.0	-	-	-	1614	1829	1945
		24.0	5425	79.6	1.53	-	-	-
		30.0	-	-	-	1890	2006	2119
2223	15	5.0	- *	-	1.76	-	-	-
		15.0	-	-	1.75	1901	2018	2072
		-	-	-	-	-	-	-

V-136* P_c data not recorded

TABLE V-14

(Continued)

INJECTOR TU-6A PERFORMANCE DATA

COLUMBIUM CHAMBER - RADIATION COOLED

Data corrected for throat expansion and contraction ratio (P_c pickup location)

Run No. (Cell B-2)	Run Duration (sec)	Data P_T (sec)	c^* corr (ft/sec)	P_c (psia)	O/F	Nozzle Min.	Temperature Ave.	(° F) Max.
2224	15	5.0	5347	80.8	1.75			
		10.0	5331	80.2	1.74			
		15.0	5319	79.7	1.75	1858	2031	2113
2225	15	5.0	5327	80.4	1.74			
		10.0	5332	79.8	1.73			
		15.0	5339	79.4	1.73	1837	2021	2104
2226	15	5.0	5378	82.1	1.60			
		10.0	5380	81.4	1.60			
		15.0	5365	81.0	1.60	1853	2024	2120
2228	15	5.0	5354	80.8	1.77			
		10.0	5340	80.2	1.77			
		15.0	5340	79.9	1.77	1739	2009	2114
2229	15	5.0	5342	80.6	1.77			
		10.0	5336	80.1	1.77			
		15.0	5332	79.8	1.77	1758	2012	2114
2230	15	5.0	5400	81.2	1.63			
		10.0	5400	80.7	1.62			
		15.0	5403	80.4	1.62	1922	2045	2127
2231	15	5.0	5396	81.0	1.62			
		10.0	5392	80.5	1.62			
		15.0	5391	80.2	1.62	1911	2039	2123
2232	15	5.0	5420	84.1	1.53			
		10.0	5415	83.6	1.53			
		15.0	5420	83.4	1.53	1908	2044	2149
2233	15	5.0	5408	83.8	1.53			
		10.0	5410	83.5	1.53			
		15.0	5413	83.4	1.53	1908	2052	2155

TABLE V-14
(Continued)

INJECTOR TU-6A PERFORMANCE DATA

COLUMBIUM CHAMBER - RADIATION COOLED

Data corrected for throat expansion and contraction ratio (P_c pickup location)

Run No. (Cell B-2)	Run Duration (sec)	Data P_T (sec)	c^* corr (ft/sec)	P_c (psia)	O/F	Nozzle Min.	Temperature Ave.	(° F) Max.
2234	15	5.0	5451	82.3	1.43			
		10.0	5459	82.0	1.43			
		15.0	5449	81.7	1.43	1898	2033	2140
2235	15	5.0	5438	82.3	1.44			
		10.0	5452	81.8	1.44			
		15.0	5453	81.5	1.43	1893	2035	2145
2236	15	5.0	5446	81.7	1.32			
		10.0	5462	81.4	1.31			
		15.0	5461	81.1	1.31	1868	2015	2130
2237	15	5.0	5444	81.6	1.31			
		10.0	5457	81.4	1.32			
		15.0	5444	81.1	1.32	1853	2015	2137
2238	30	5.0	5383	80.6	1.61			
		10.0	5389	80.3	1.61			
		15.0	5391	80.0	1.61	1872	2040	2160
		25.0	5371	79.6	1.61			
		30.0	5364	79.5	1.61	2045	2220	2342

To preclude any forced convective cooling of the chamber during firings, both ends of the shell assembly were sealed in a manner such that only a small gap existed between the shell end plates and the chamber. The narrow annulus (gap) was required to allow thermal expansion of the columbium chamber during firings.

Injector TU-3B was selected for use in the adiabatic wall demonstration test series. The exceedingly low heat rejection characteristics (approximately $0.9 \text{ BTU/sec/in.}^2$) of this injector together with improved "hot wall" performance made this injector the most likely candidate for the first adiabatic wall tests. Three firings were made, the first being a 5-second duration performance check run. The second firing was a 60-second duration run at a 1.55 mixture ratio with an average nozzle temperature below 2000°F . The third test was a 120-second duration firing at a 1.32 mixture ratio. The mixture ratio was shifted to a more fuel rich condition since previous tests indicated TU-3 injector tended to yield higher performance at lower mixture ratio operation. The test produced a c^* increase of 2 percent while the average nozzle temperature remained slightly below 2000°F . Average temperature profiles for the third (120-second) test are presented in Figure V-133. It will be seen that the chamber and nozzle temperatures stabilized during the duration of the test while the temperature of the shell and the injector did not. There was also a very gradual rise in the insulation temperature. In other words, the system had not reached a thermal equilibrium condition. The range of temperatures measured at the 120-second point of the run for each of the assembly components is shown in Figure V-134. A summary of the performance data from each of the three firings is presented in Table V-15. A post-fire examination of the columbium chamber revealed that the chamber internal oxidation resistant coating remained in excellent condition.

b. Injector TU-6A - Columbium Chamber

This injector was tested in the columbium chamber S/N 2 using the same insulated shell assembly filled with alumina bubbles as described above.

Three test firings were conducted in this "buried" configuration consisting of a 5 second, 30 second and 240 second duration test. The same chamber thermocouple installation was used as during the radiation cooled fire tests (24 thermocouples) with four thermocouples

TABLE V-15
MODEL 8374
TEST DATA SUMMARY
ADIABATIC WALL THRUST CHAMBER TESTS
Injector TU-3 Test Assy 8374-470060
Columbium Chamber S/N 1

Run No.	L.O.R. sec	Data Pt. sec	$\dot{W}_F$ lb/sec	$\dot{W}_{ox}$ lb/sec	$\dot{W}_{total}$ lb/sec	O/F Ratio	P_c psia	c^*_{cor} ft/sec	Avg Nozzle Temp. °F
2141	10.5	10.0	0.140	0.216	0.356	1.54	79.8	5025	1643
2142	61.0	10.0	0.140	0.218	0.358	1.55	79.3	4960	1569
		30.0	0.140	0.217	0.357	1.55	78.7	4940	1917
		60.0	0.139	0.218	0.357	1.56	78.5	4950	1944
2143	121.0	10.0	0.147	0.197	0.344	1.33	77.7	5065	1607
		30.0	0.147	0.195	0.342	1.32	76.9	5060	1901
		60.0	0.147	0.194	0.341	1.32	76.8	5065	1922
		90.0	0.147	0.193	0.340	1.31	76.4	5050	1915
		120.0	0.147	0.194	0.341	1.32	76.3	5040	1919

inserted in the alumina bubbles at 1-1/4 in. from the chamber wall. A water-cooled coil was attached to the injector flange to cool the chamber-injector velbestos gasket. The temperature profile for the 4 minute "buried" test is shown in Figure V-135 with three views of the test setup shown in Figures V-136, V-137, and V-138.

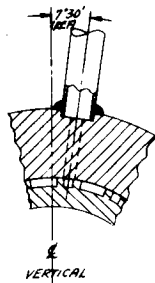
A tabulation of the test results is given in Table V-16. The throat temperature was stabilized at a maximum of 2701° F with an average temperature of 2644° F.

The results of these tests show the feasibility of the insulated concept and the resultant low driving gas temperature and high performance of this unbalanced injector.

TABLE V-16
INJECTOR TU-6A - COLUMBIUM CHAMBER
"BURIED" TESTS

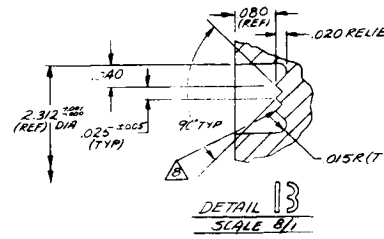
(Data corrected for throat expansion and contraction ratio)

Run No. (Cell B-2)	Run Dur. (sec)	Data Pt (sec)	c* _c (ft/sec)	P _c (psia)	O/F	Nozzle Temp. (° F)		
						Min.	Ave.	Max.
2239	5	5.0	5432	79.9	1.59	-	-	-
2240	30	5.0	5418	79.9	1.60	-	-	-
		15.0	5438	79.9	1.61	-	-	-
		30.0	5411	79.6	1.62	2132	2313	2482
2241	240	30.0	5395	79.2	1.63	1992	2310	2470
		60.0	5388	79.1	1.63	2330	2514	2636
		120.0	5367	78.9	1.64	2550	2610	2678
		180.0	5352	78.8	1.64	2582	2636	2694
		240.0	5350	78.6	1.64	2592	2644	2701



SECTION 11-11

CSK .220" $\pm$.002 DIA X 120" $\pm$.5"
 .196" $\pm$.002 DIA 8 HOLES EQUALLY
 SPACED WITHIN .005
 .25 DEEP
 MS124675 INSERT AREA

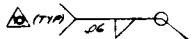


DETAIL 13
 SCALE 8/1

FLARED TUBE END PER MS 3350*

AN 818-AC COUPLING
 MS 20819-AC SLEEVE
 16 EACH REQD

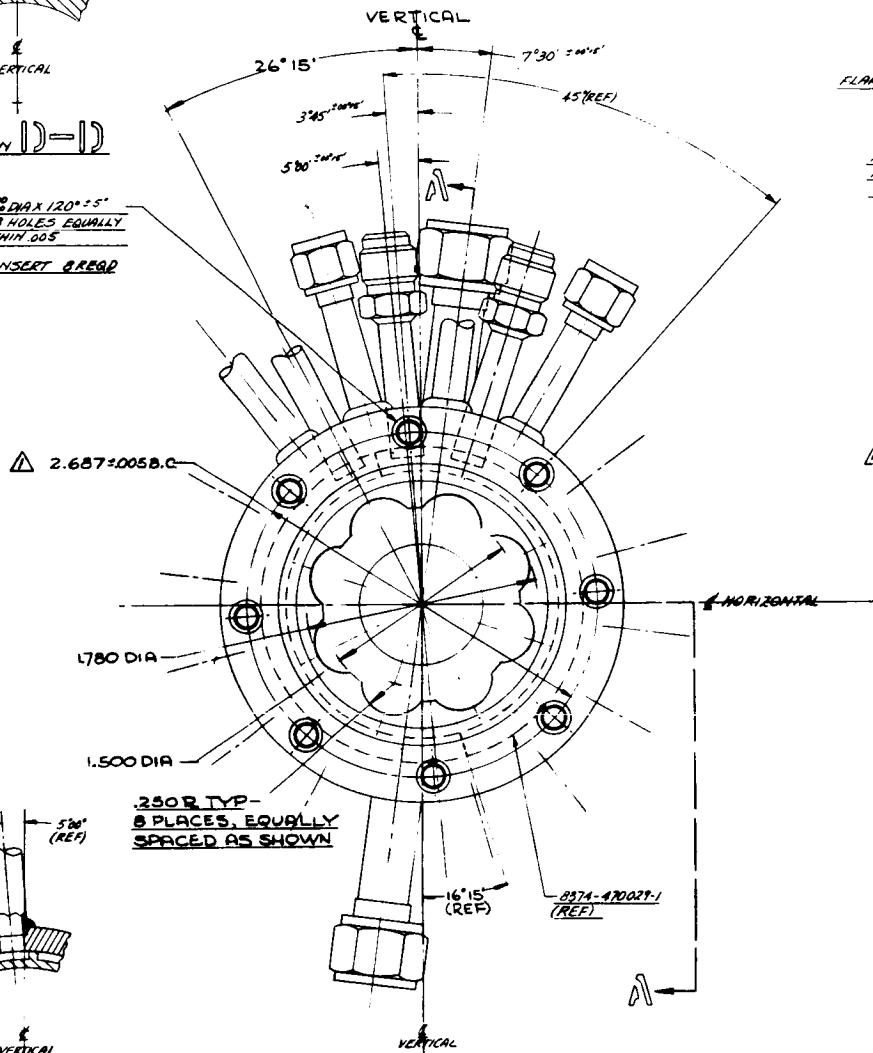
16 REQD -9



8374-470027-1

SEE DETAIL 13

ELECTRON BEAM
 WELD, FULL PENETRATION
 .140 MAX DEPTH
 MACHINE WELD FLUSH
 AFTER WELDING.



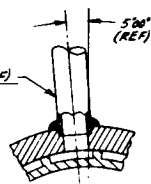
2.657 $\pm$.005 B.C.

1780 DIA

1.500 DIA

.250 B TYP -
 8 PLACES, EQUALLY
 SPACED AS SHOWN

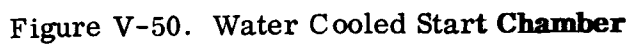
8374-470031-1 (REF)



SECTION 12-12

8374-470027-1 (REF)

V-142-1



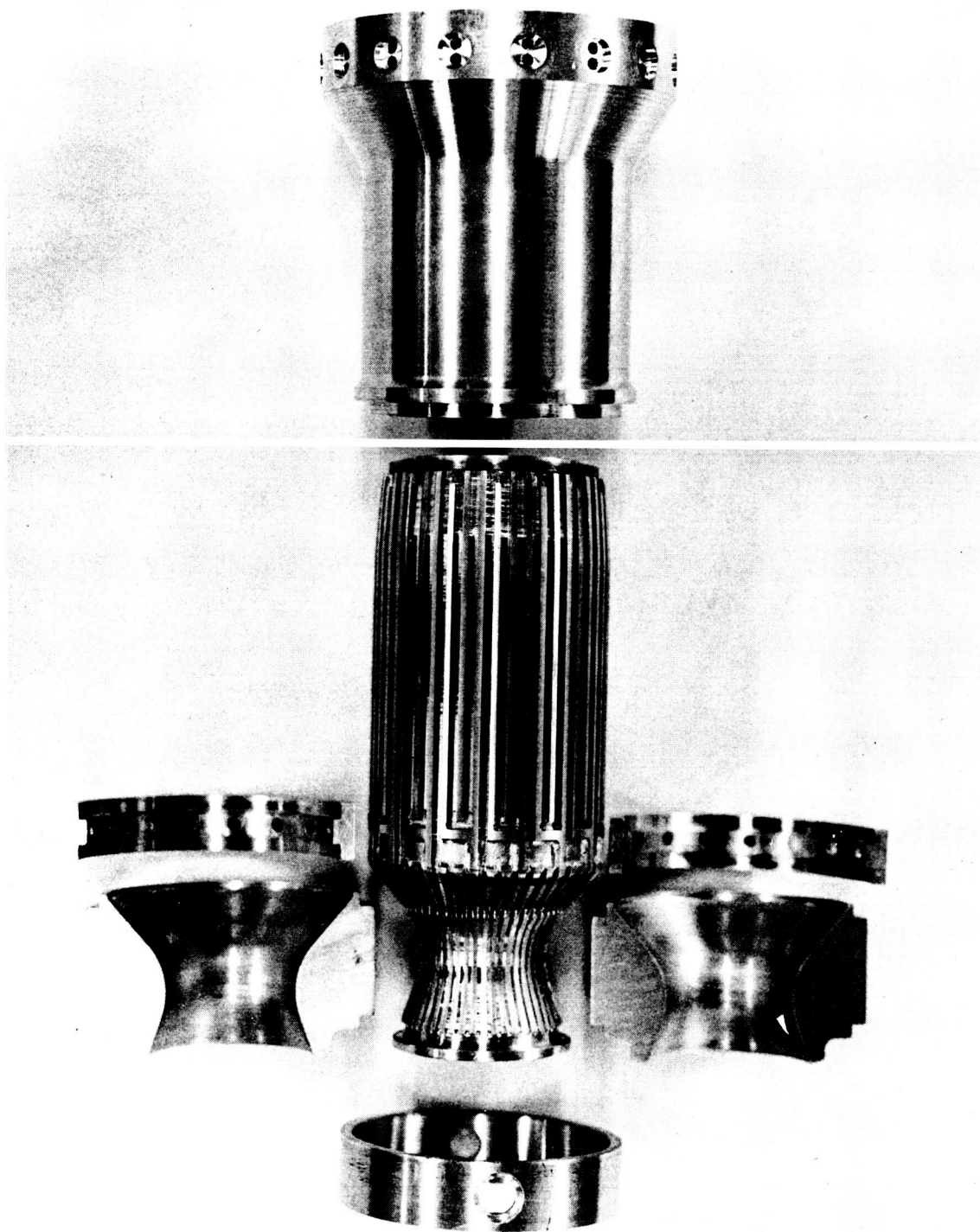


Figure V-51. H_2O Cooled Chamber

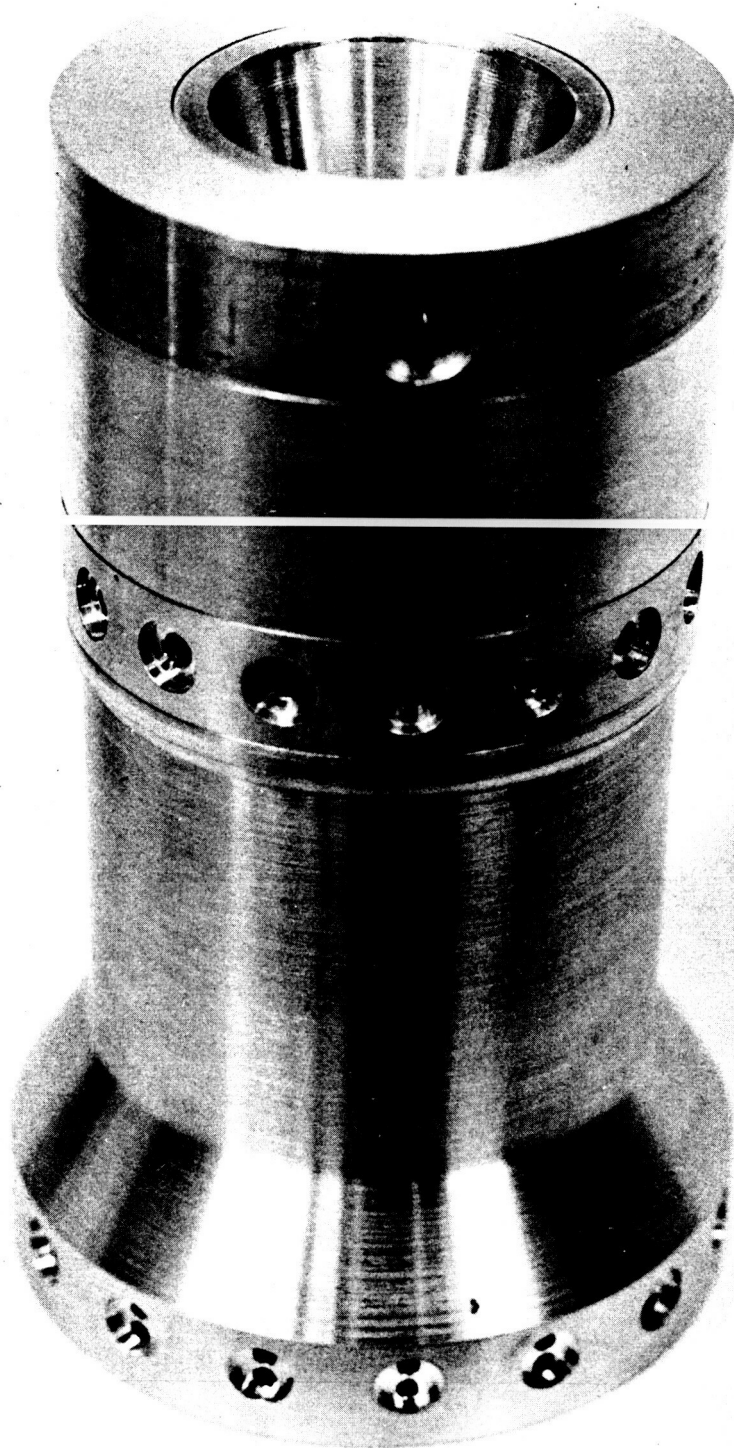


Figure V-52. H_2O Cooled Chamber

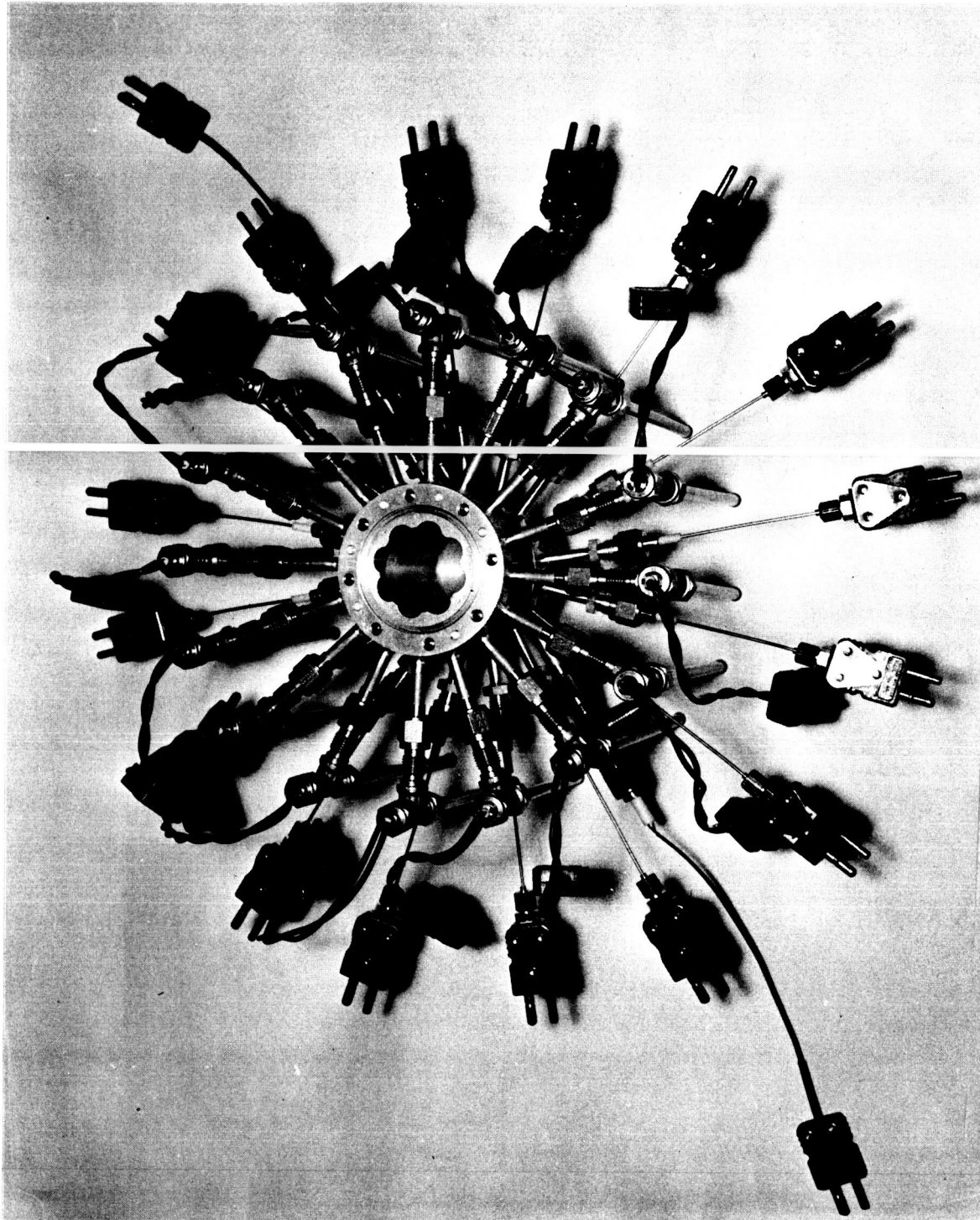


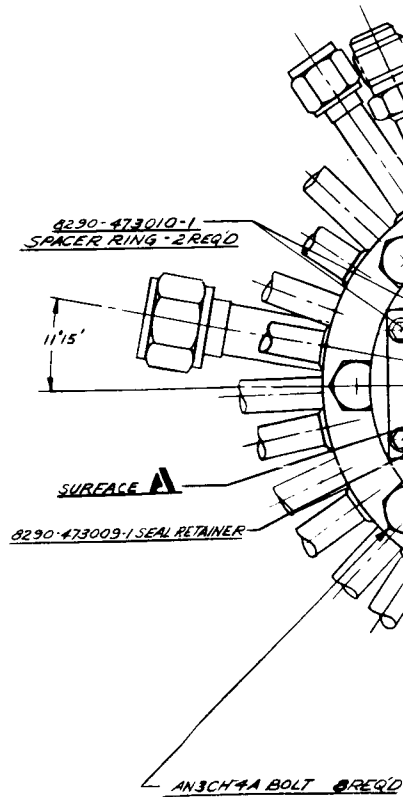
Figure V-53. H_2O Cooled Chamber with Thermocouples

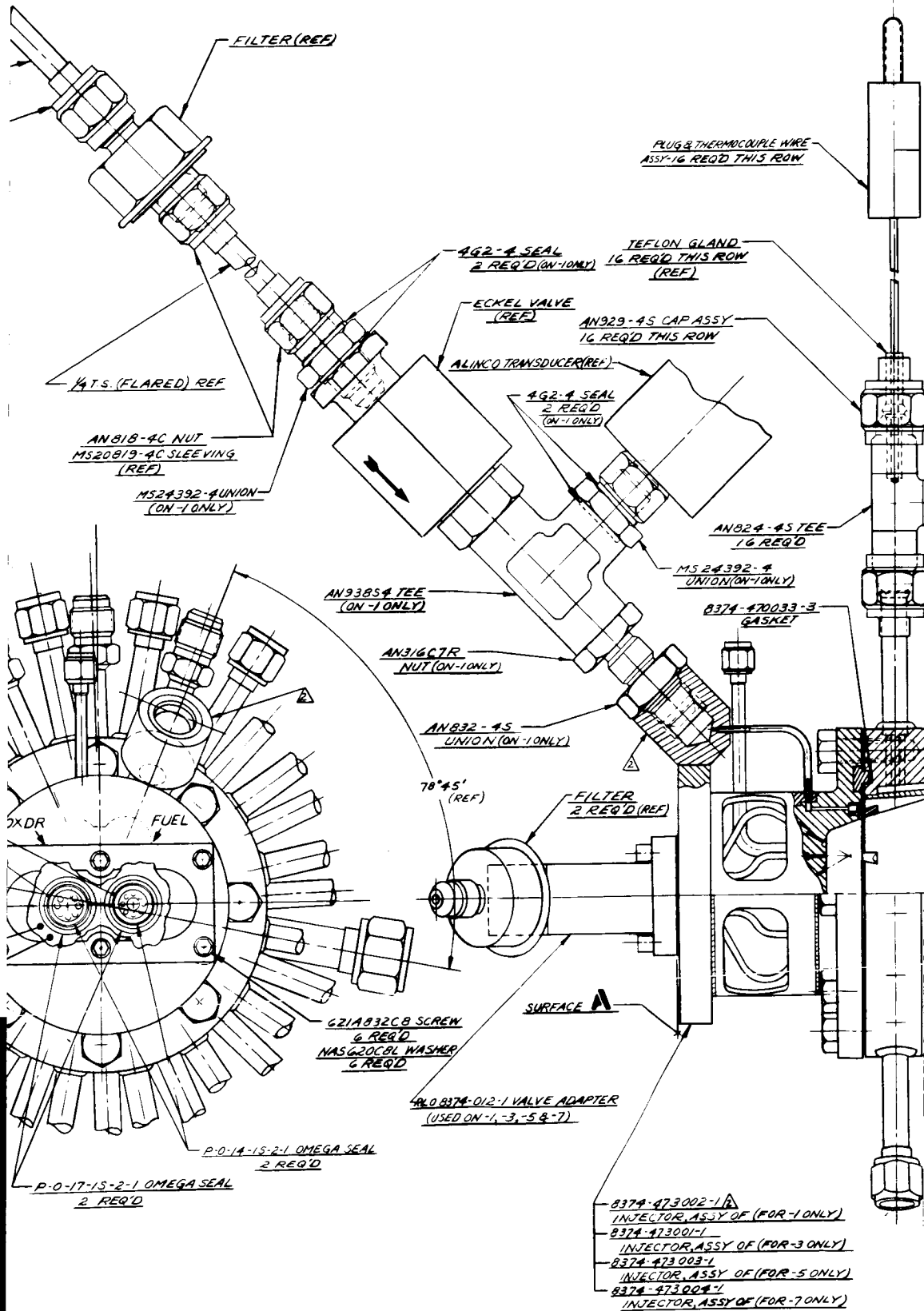
1/4 T.S. (REF)

ANB18-4C NUT
MS20819-4C SLEEVING
(REF)

- △ INJECTOR SHOWN ON FACE OF DRAWING WITH BOSS FOR MOUNTING VALVE FILTER & TRANSDUCER IS CONFIGURATION OF 8172-473002-1 INJECTOR ONLY.
- △ FOR SURFACE THERMOCOUPLE TO BE MOUNTED ON ALUMINUM:
APPLY WELD BEAD OF 4043 AL ALY APPROX .250 TO .375 DIA X .125 TO .148 HIGH. DRILL .062 DIA HOLE THRU EACH BEAD PARALLEL & APPROX .091 FROM SURFACE SHOWN. DIRECTION OF HOLE IN BEAD OPTIONAL. THERMOCOUPLES SHALL BE 22 GAUGE CHROMEL-ALUMEL THERMOCOUPLE WIRES WITH TEFLON/GLASS INSULATION. ATTACH AT EACH WELD BEAD BY PERMING. THE THERMOCOUPLE JUNCTION SHALL BE CONFINED TO THE BONDED AREA. THE LEAD WIRES & WELD BEAD SHALL BE POTTED WITH A SUITABLE RESILIENT POTTING COMPOUND TO PREVENT WIRE BREAKAGE. LEAD WIRES SHALL BE APPROX. 4.0 INCHES LONG.

NOTES:





V-146-2

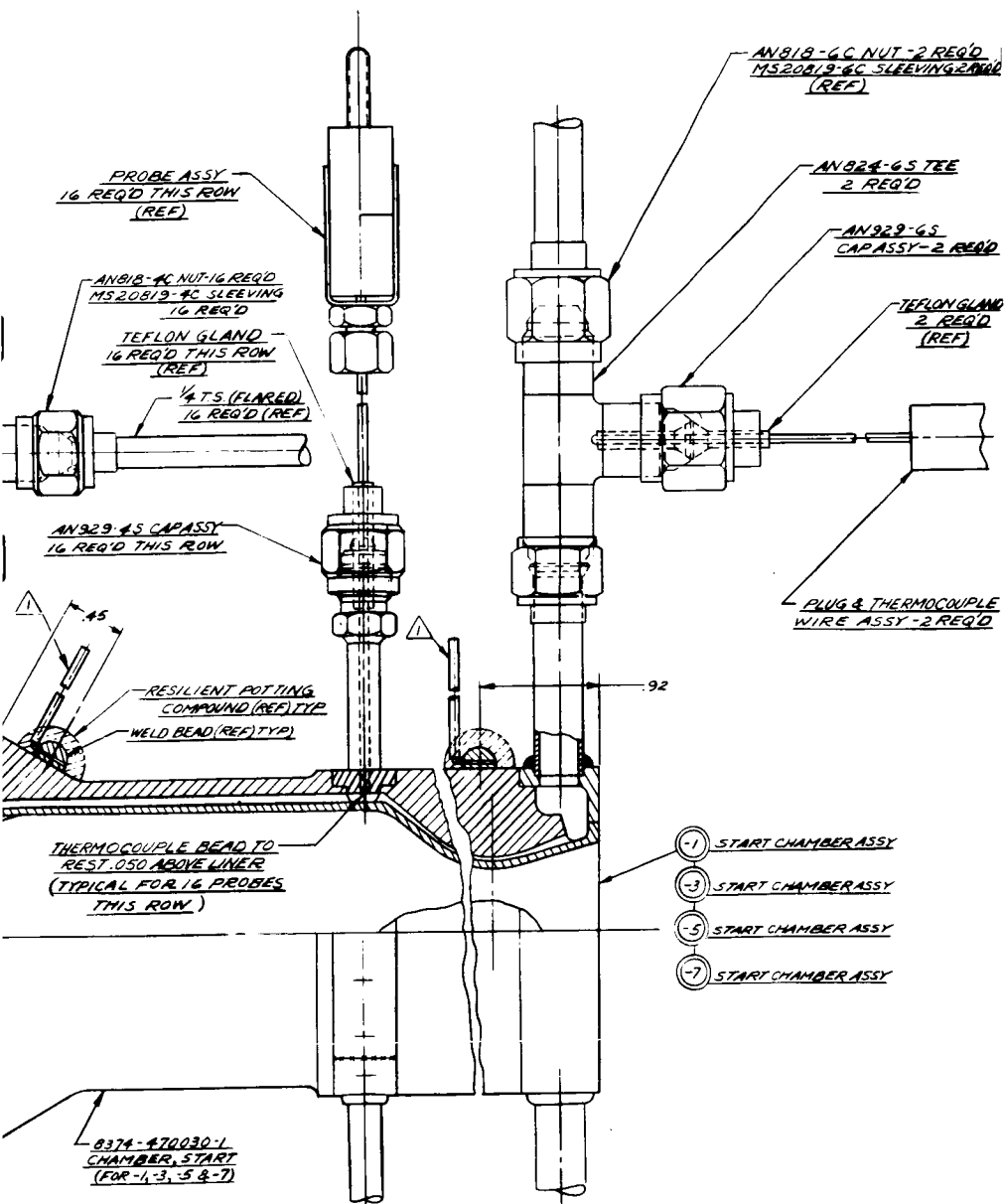
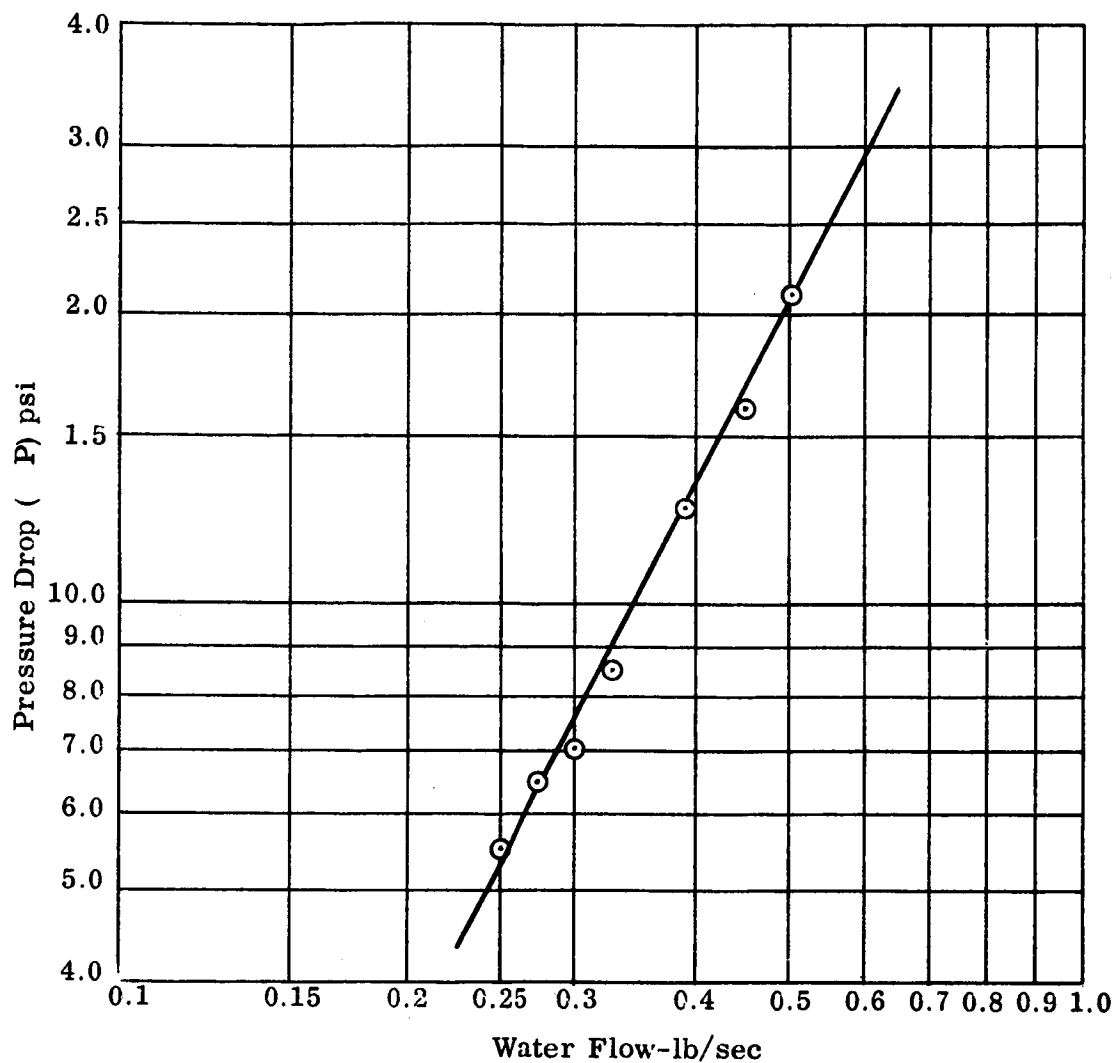
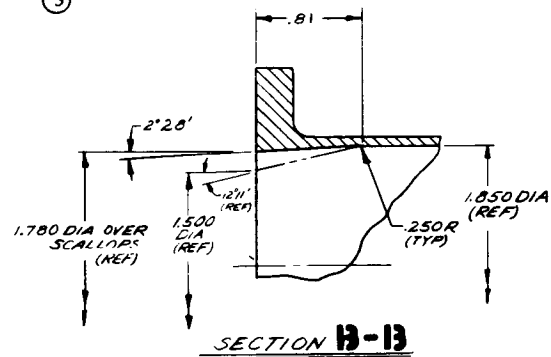
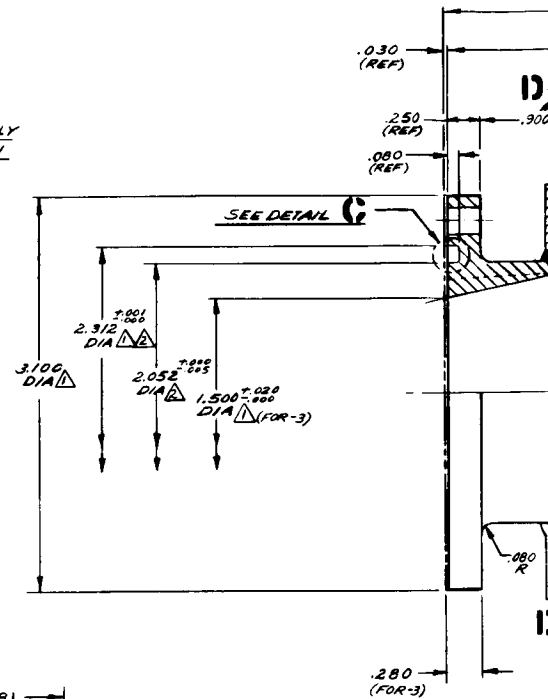
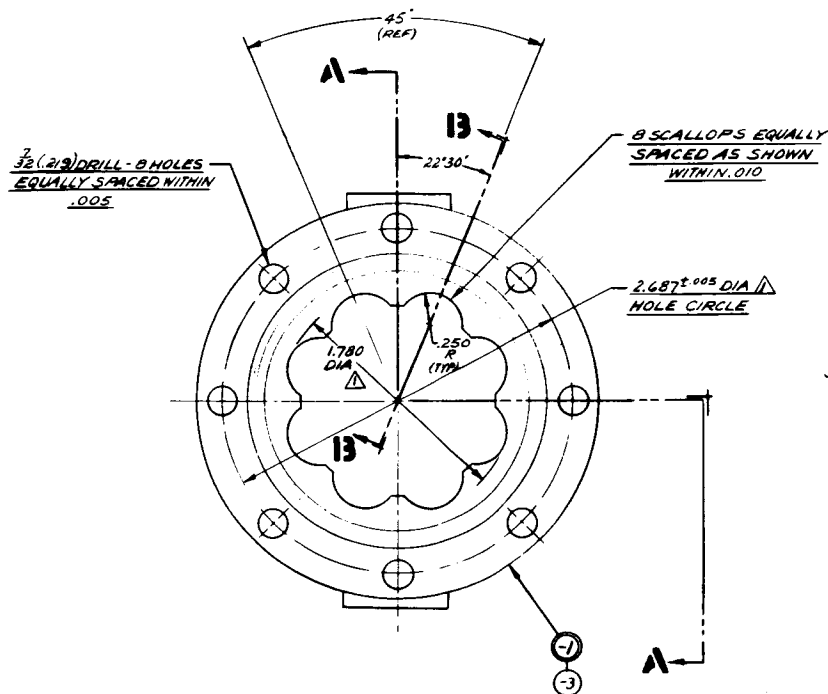
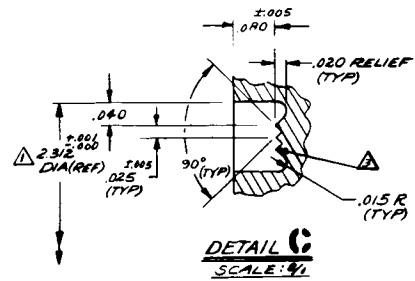
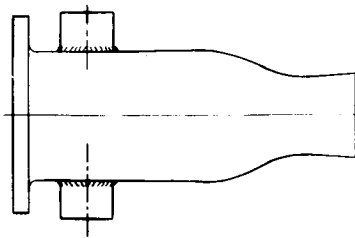


Figure V-54. H₂O Cooled Start Chamber



Notes: All Thermocouples Installed
 Nominal Total Flow Rate = 0.418 lb H₂O/sec
 Flow Stand No. 1
 Stand Orifice No. 68 & No. 1

Figure V-55. Pressure Drop Data



- △ CLEAN PER BPS
 - △ PRIOR TO MACH.
 - △ THIS DIMEN.S
 - △ CREST OF SERR
 - △ EDGE TO .005
 - △ THESE DIAMETER
 - △ THESE DIAMETER
- NOTES:

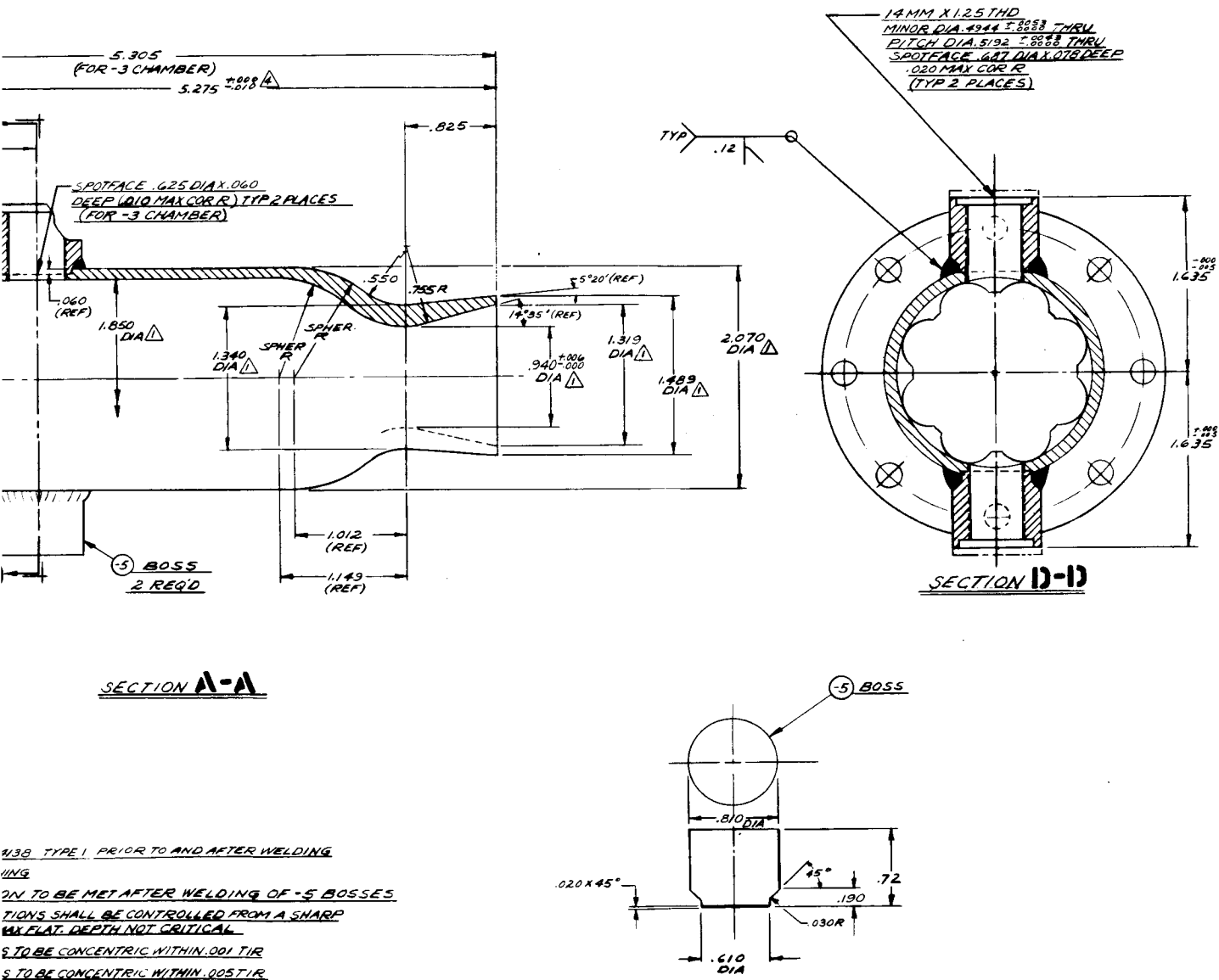


Figure V-56. Stainless Steel Chamber

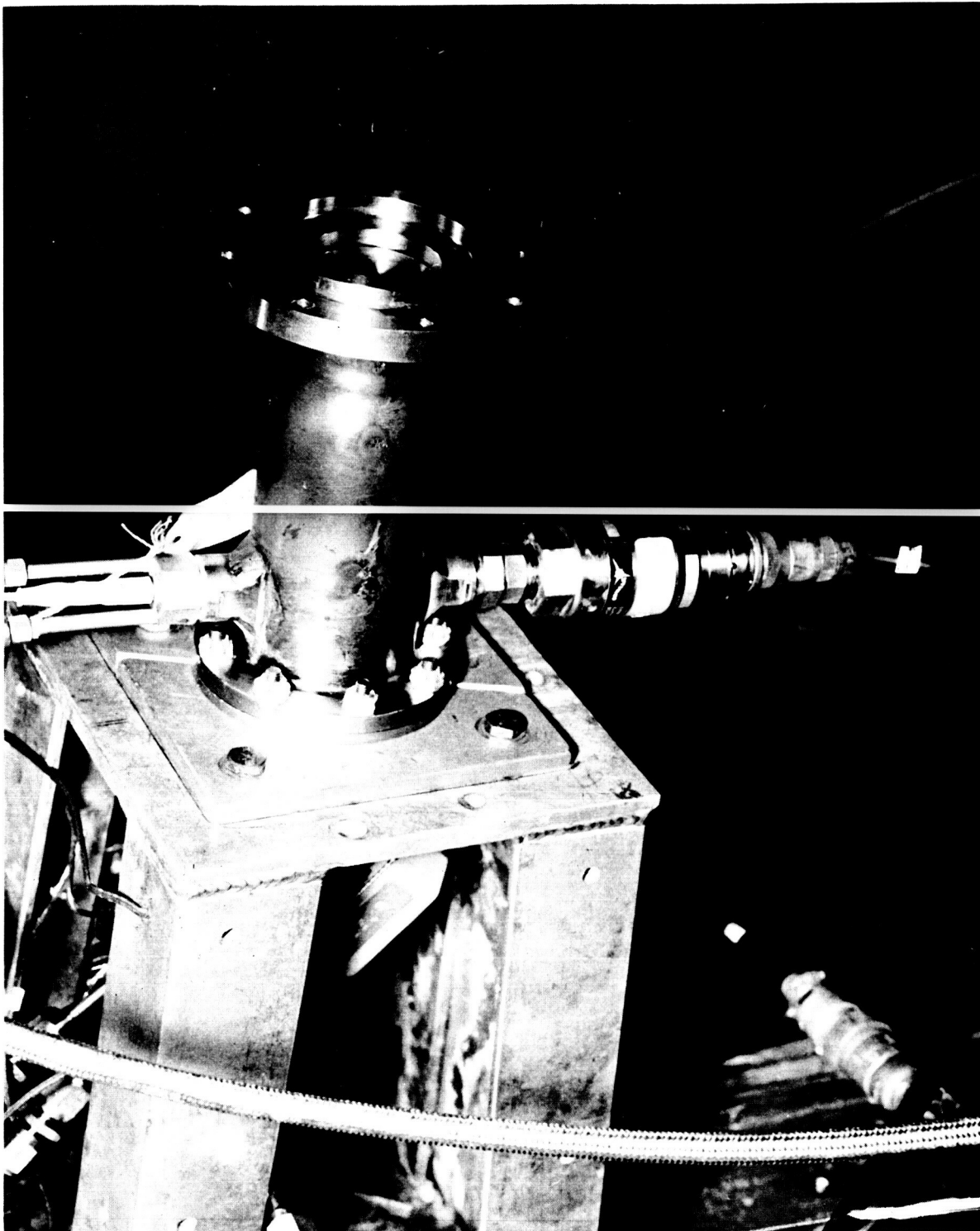
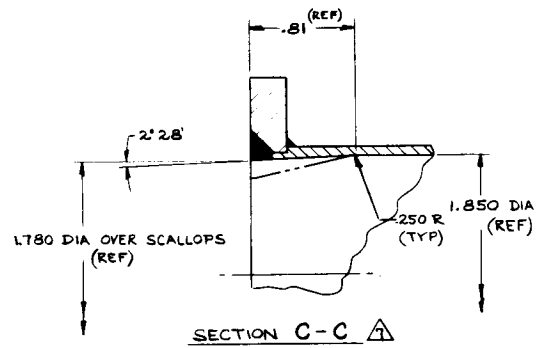
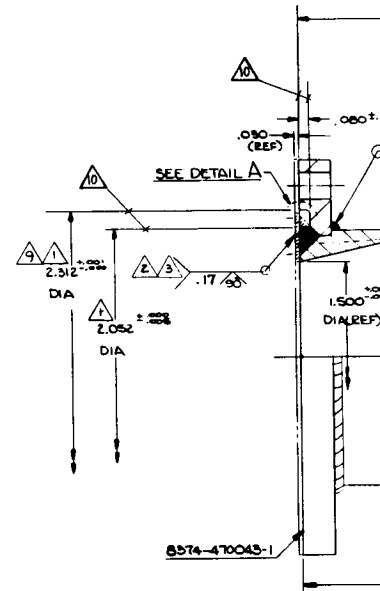
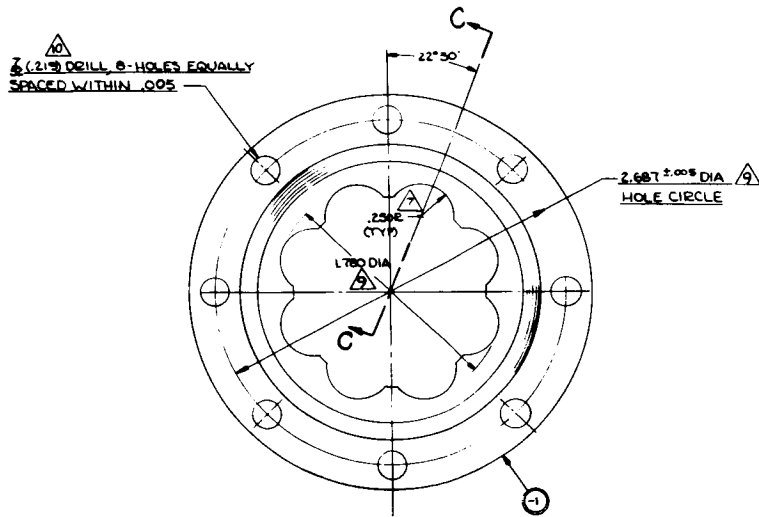
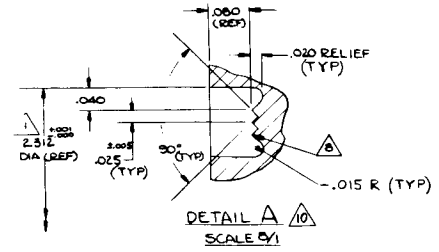
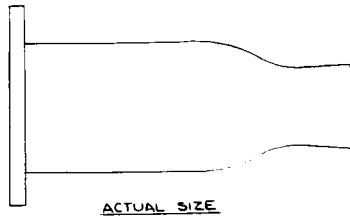
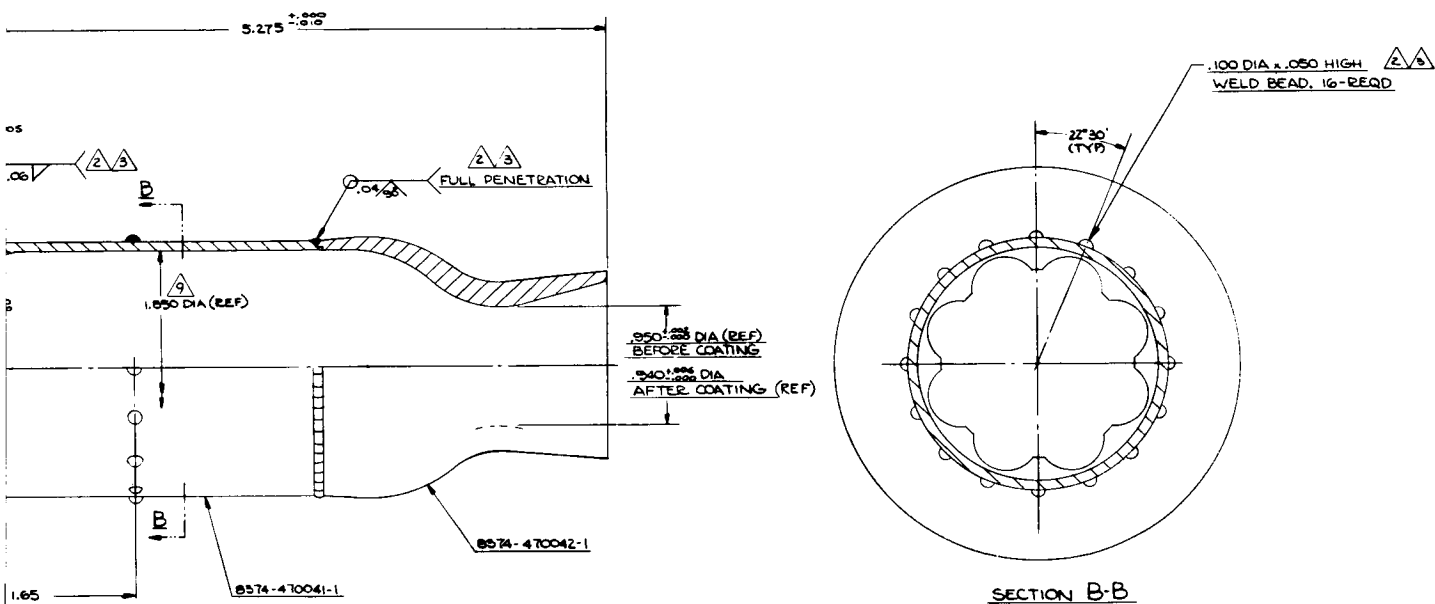


Figure V-57. Stainless Steel Chamber





- 10. THESE SURFACES SHALL BE FREE OF COATING.
- 9. THESE DIAMETERS SHALL BE CONCENTRIC WITHIN .005 TIR.
- 8. CREST OF SEPARATIONS SHALL BE CONTROLLED FROM A SHARP EDGE TO .005 MAX FLAT. DEPTH NOT CRITICAL.
- 7. 250R SCALLOPS ARE TO BE MACHINED BEFORE COATING.
- 6. SYLOR 2500C SILICIDE COATING TO BE APPLIED PER MPB C-64 TO ALL INTERNAL & EXTERNAL SURFACES EXCEPT AS INDICATED PER NOTE 10.
- 5. CLEAN PER BPS 4138 TYPE 2 & PACKAGE PER BPS 4266 PRIOR TO SHIPMENT TO COATING VENDOR.
- 4. DYE PENETRANT INSPECT ALL WELDS (EXCEPT THERMOCOUPLE BEADS) PER MIL-T-8866 (BPS 4089 TYPE 2). DO NOT DIP.
- 3. ALL WELDING & TACKING TO BE DONE IN A VACUUM PURGED INERT GAS CHAMBER. USE CR-12 FILLER ROD. WELDER CERTIFIED FOR TITANIUM WELDING IS ACCEPTABLE.
- 2. BEFORE FUSION WELDING, CLEAN ALL PARTS AS FOLLOWS:
 - 2.1) CLEAN PER BPS 4138 TYPE 1, USING OAKITE NO. 614 OR ALTEX CLEANER. RINSE THOROUGHLY.
 - 2.2) PICKLE FOR ONE HALF HOUR AT ROOM TEMPERATURE IN A SOLUTION OF 50 PCT NITRIC ACID, 5 PCT HYDROFLUORIC ACID & 45 PCT WATER (PCT BY VOLUME). RINSE THOROUGHLY IN RUNNING WATER & DRY.
 - 2.3) BEFORE WELDING AND AFTER FIXTURING & FIT-UP, RECLEAN LOCALLY IF NECESSARY. CLEAN WITH METHYLENE CHLORIDE FOLLOWED BY LOCAL WIRE BRUSHING WITH A CLEAN STAINLESS STEEL ROTARY BRUSH.

- 1. THESE DIAMETERS ARE TO BE CONCENTRIC WITHIN .001 T.I.R.
- NOTES

Figure V-58. Columbum Start Chamber

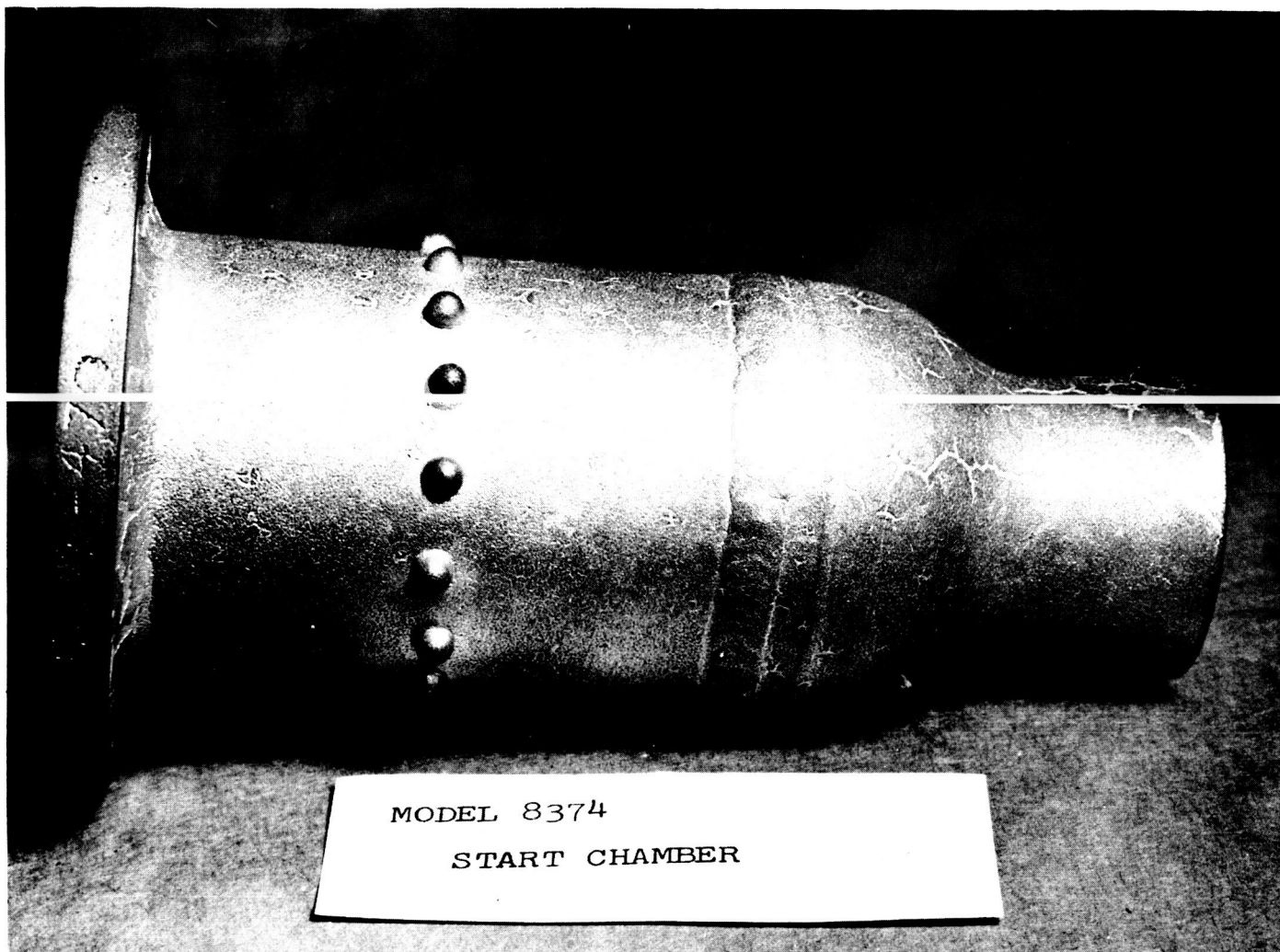


Figure V-59. Model 8374 Start Chamber

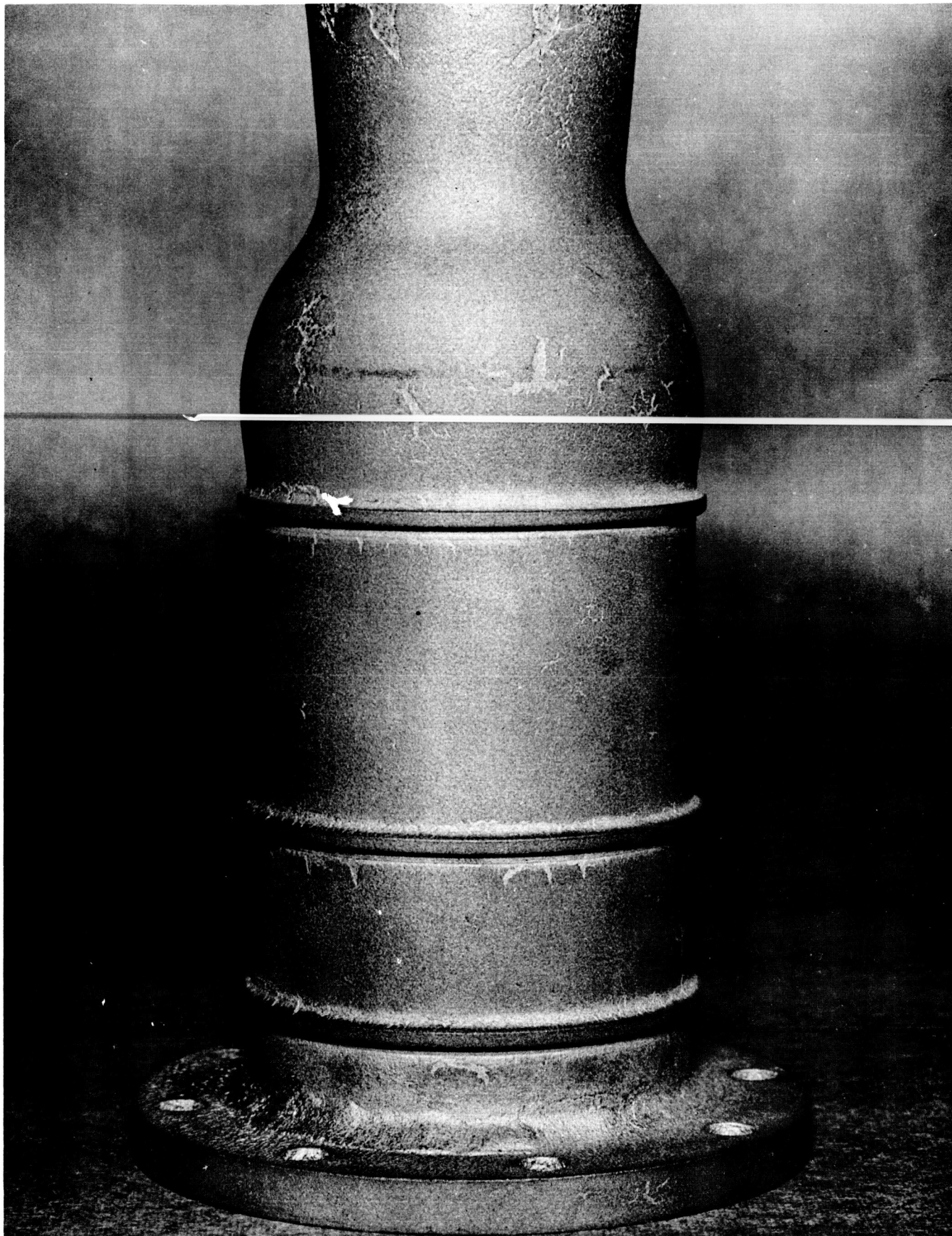


Figure V-60. Columbian Start Chamber S/N 2



Figure V-61. T-1 Injector

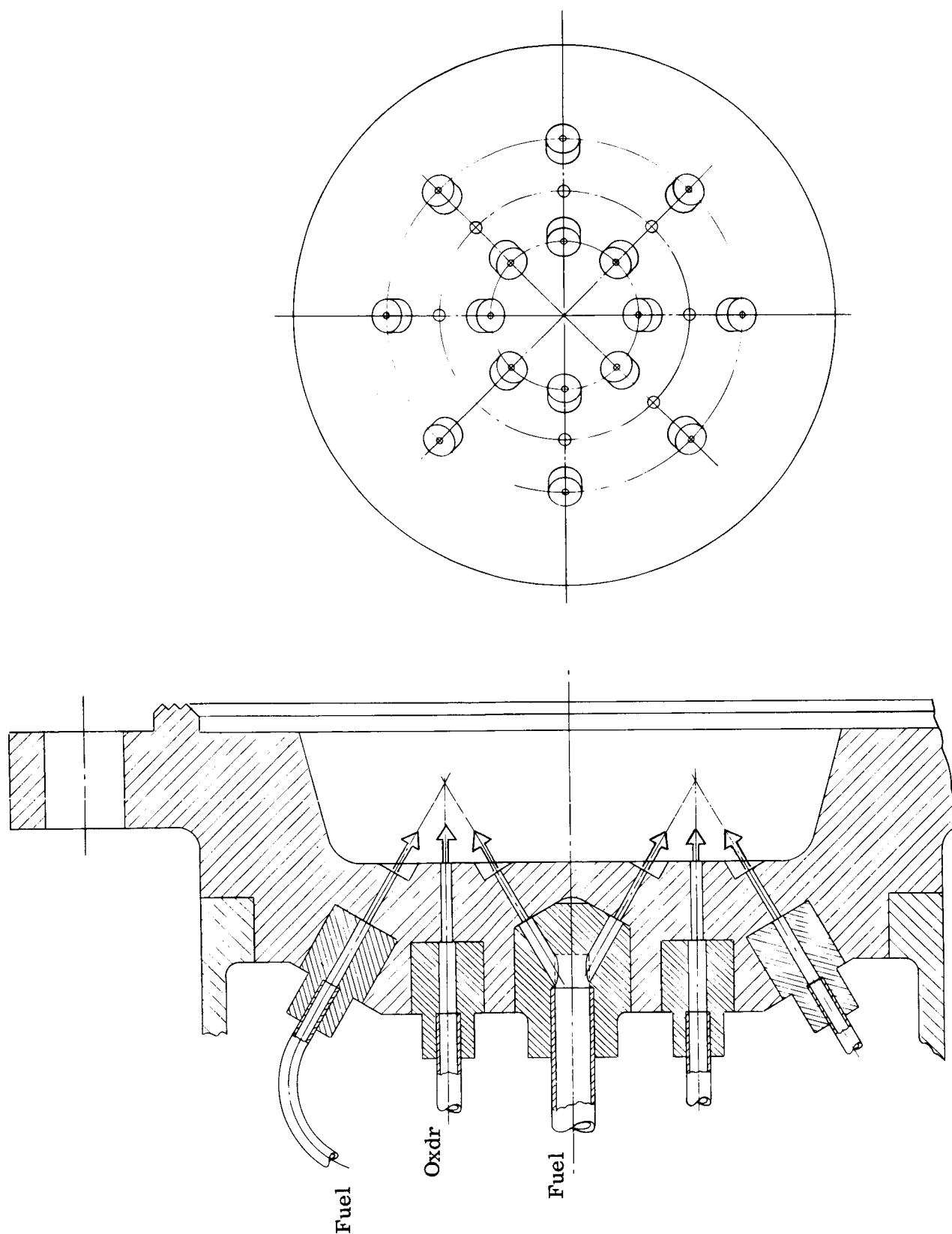


Figure V-62. 8 Triplet Balanced Injector T-1 (Test Configuration)

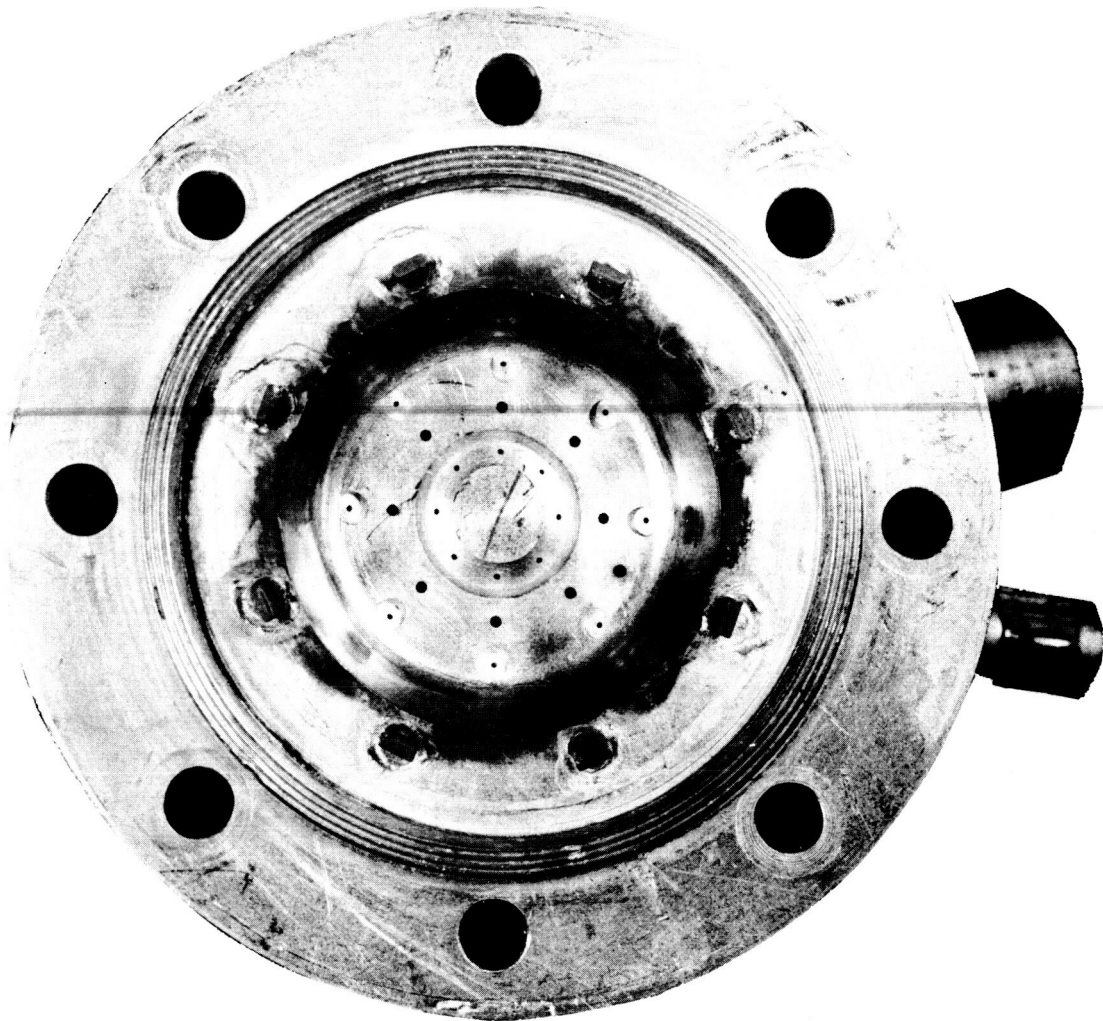


Figure V-63. TF-2 Injector

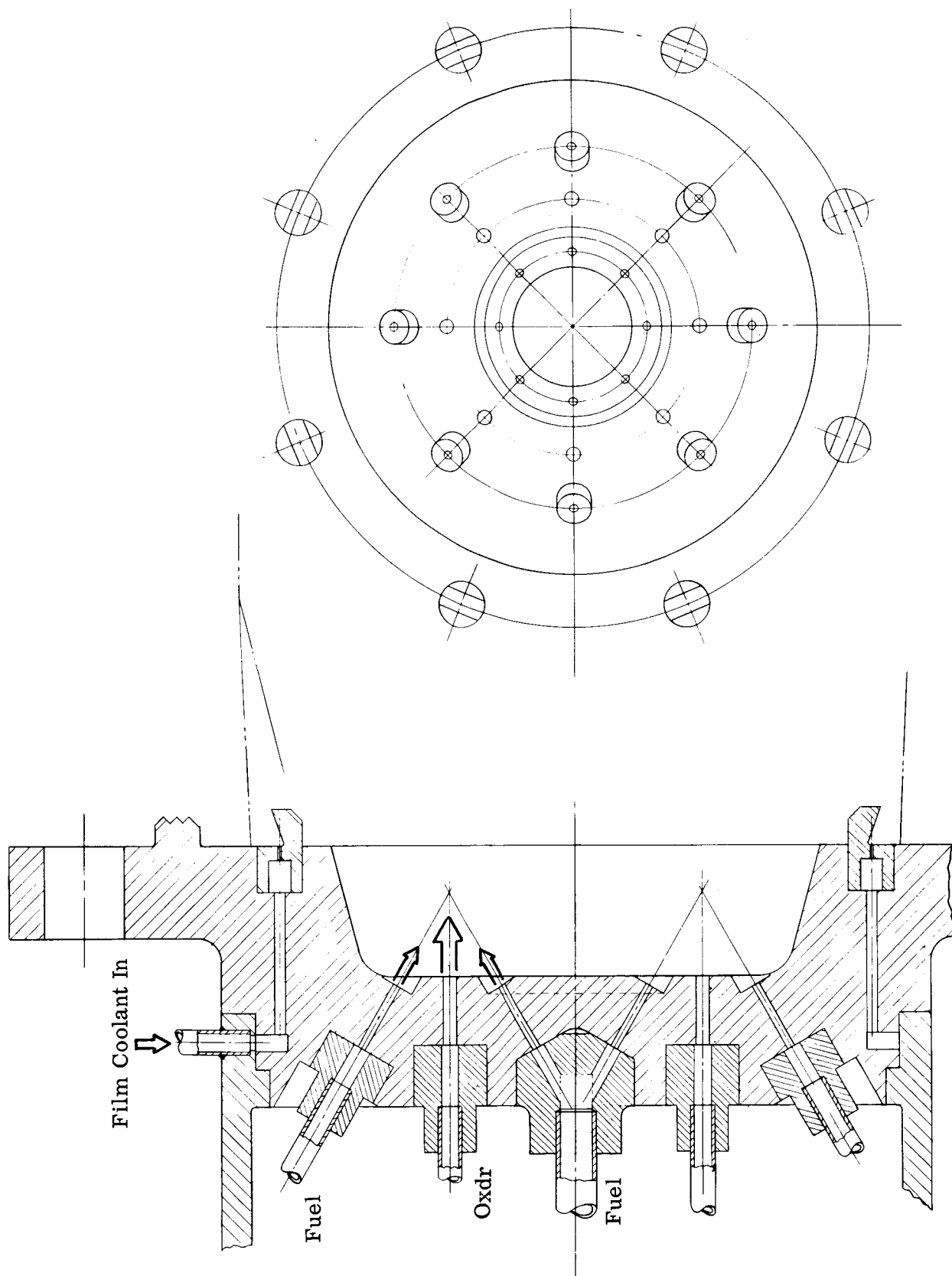


Figure V-64. 8 Triplet Film Cooled Injector TF-2 (Test Configuration)

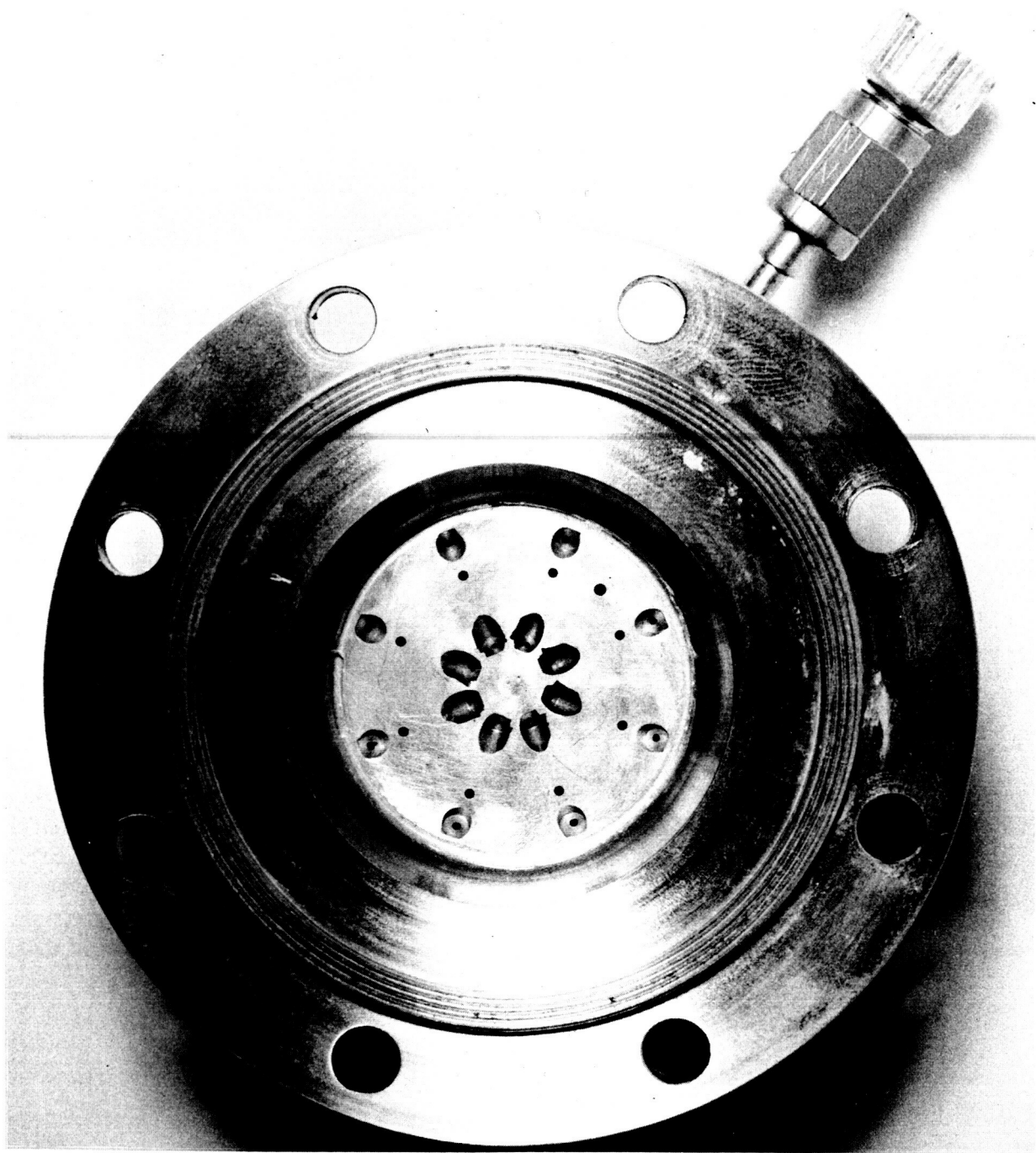
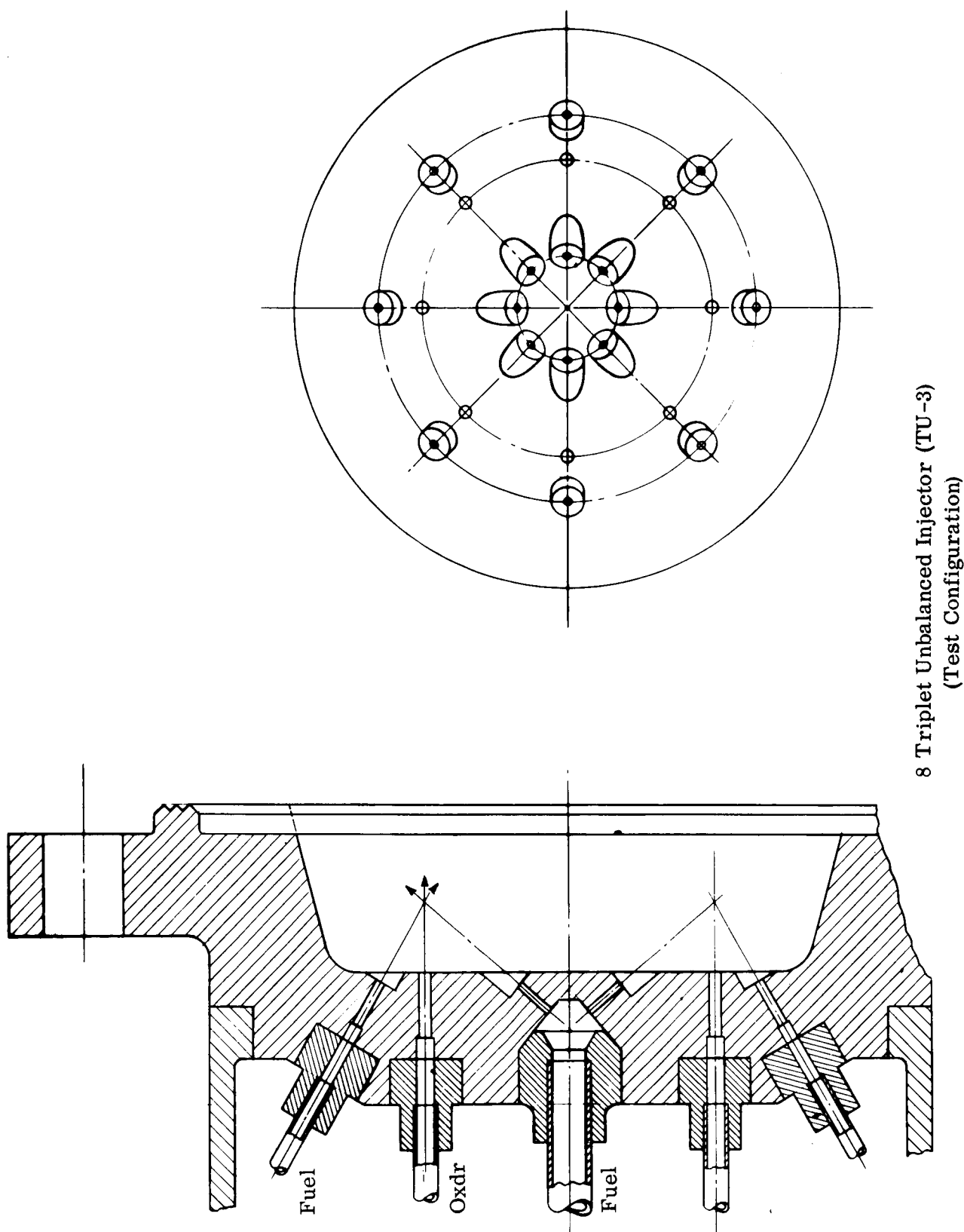


Figure V-65. TU-3 Injector



8 Triplet Unbalanced Injector (TU-3)
(Test Configuration)

Figure V-66. 8 Triplet Unbalanced Injector TU-3 (Test Configuration)

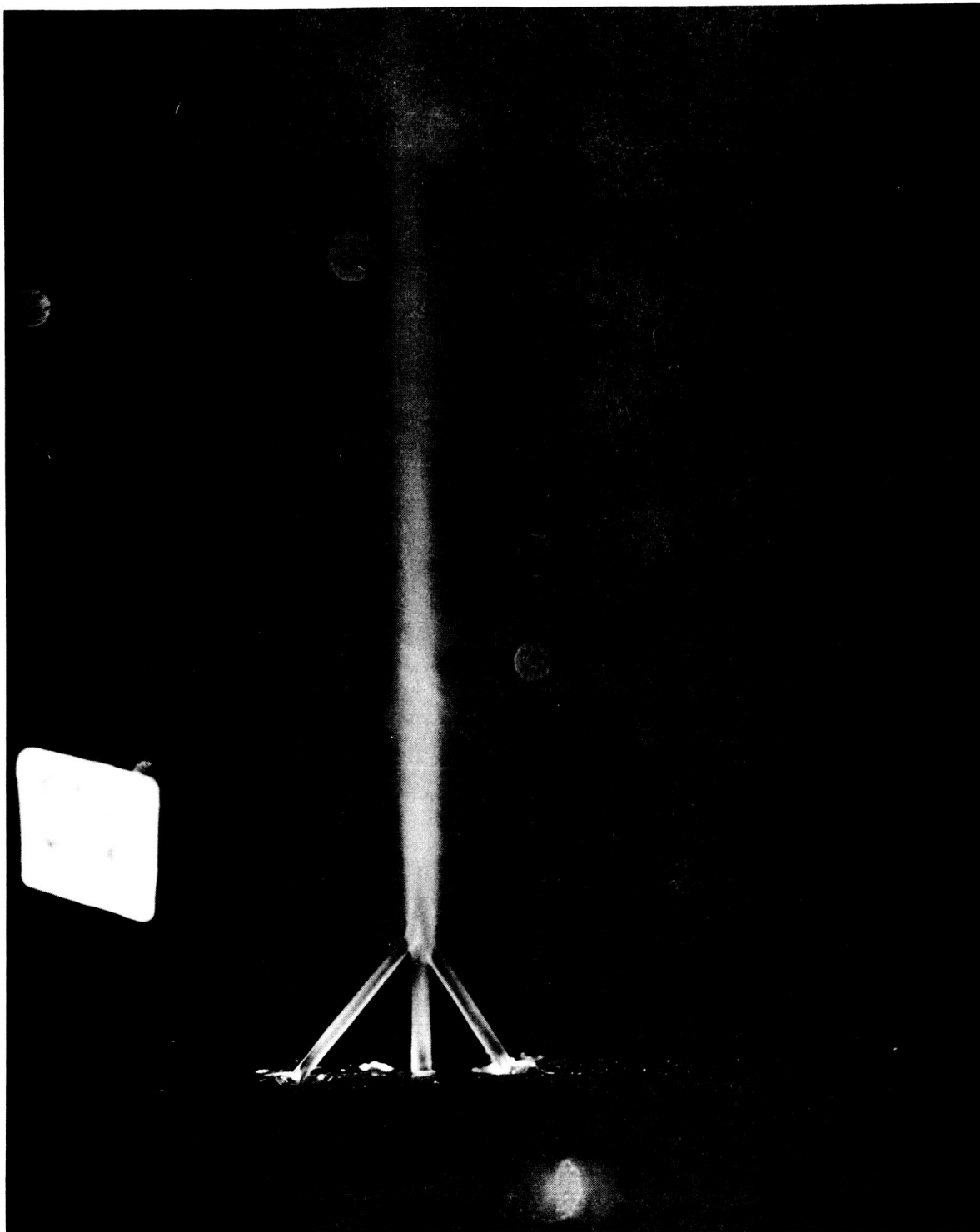


Figure V-67. H_2O Flow - Unbalanced Triplet

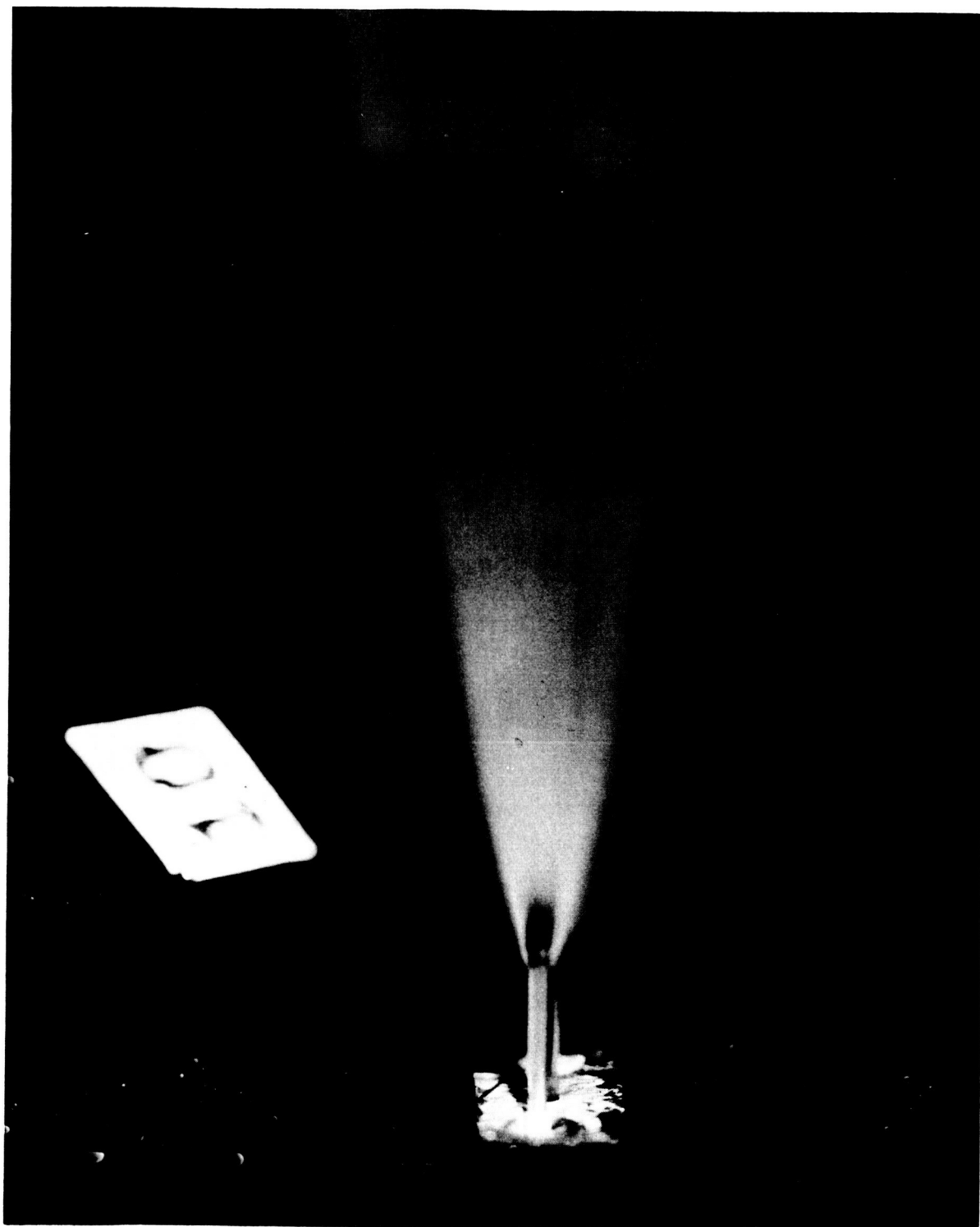


Figure V-68. H₂O Flow - Unbalanced Triplet

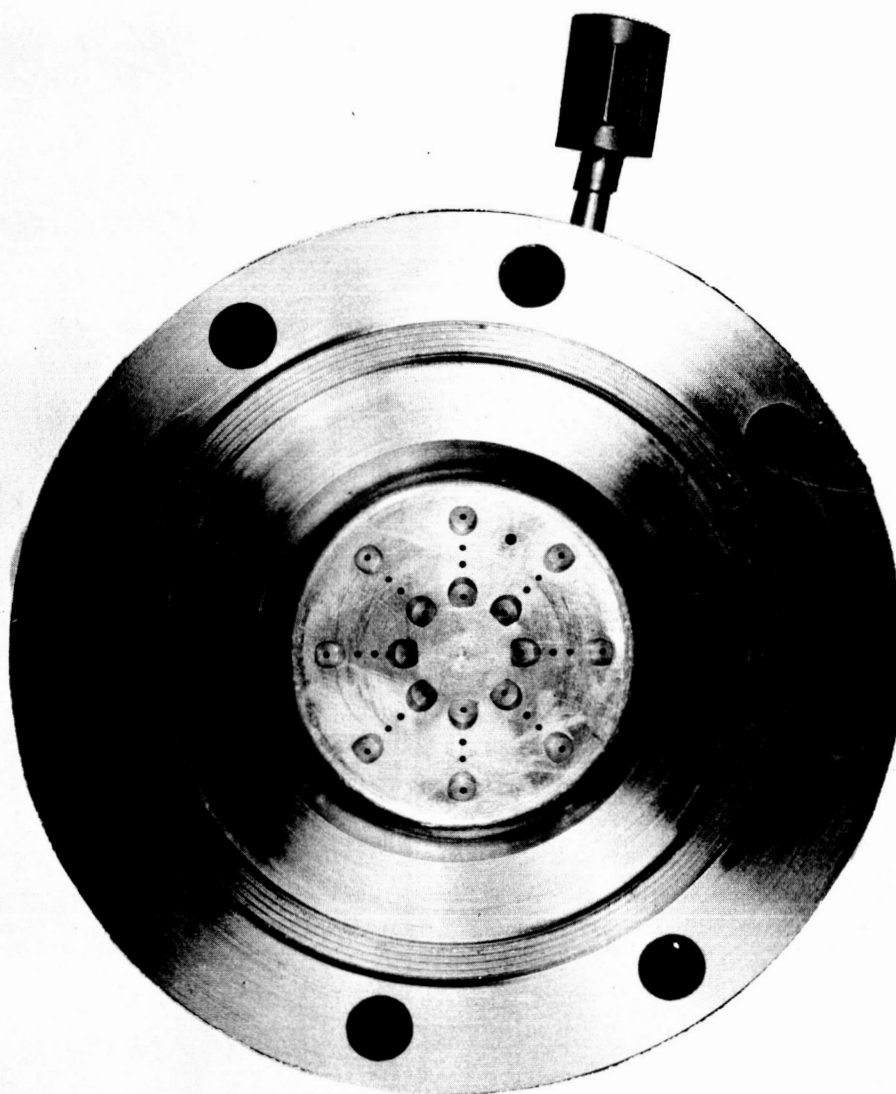


Figure V-69 TS-4 Injector

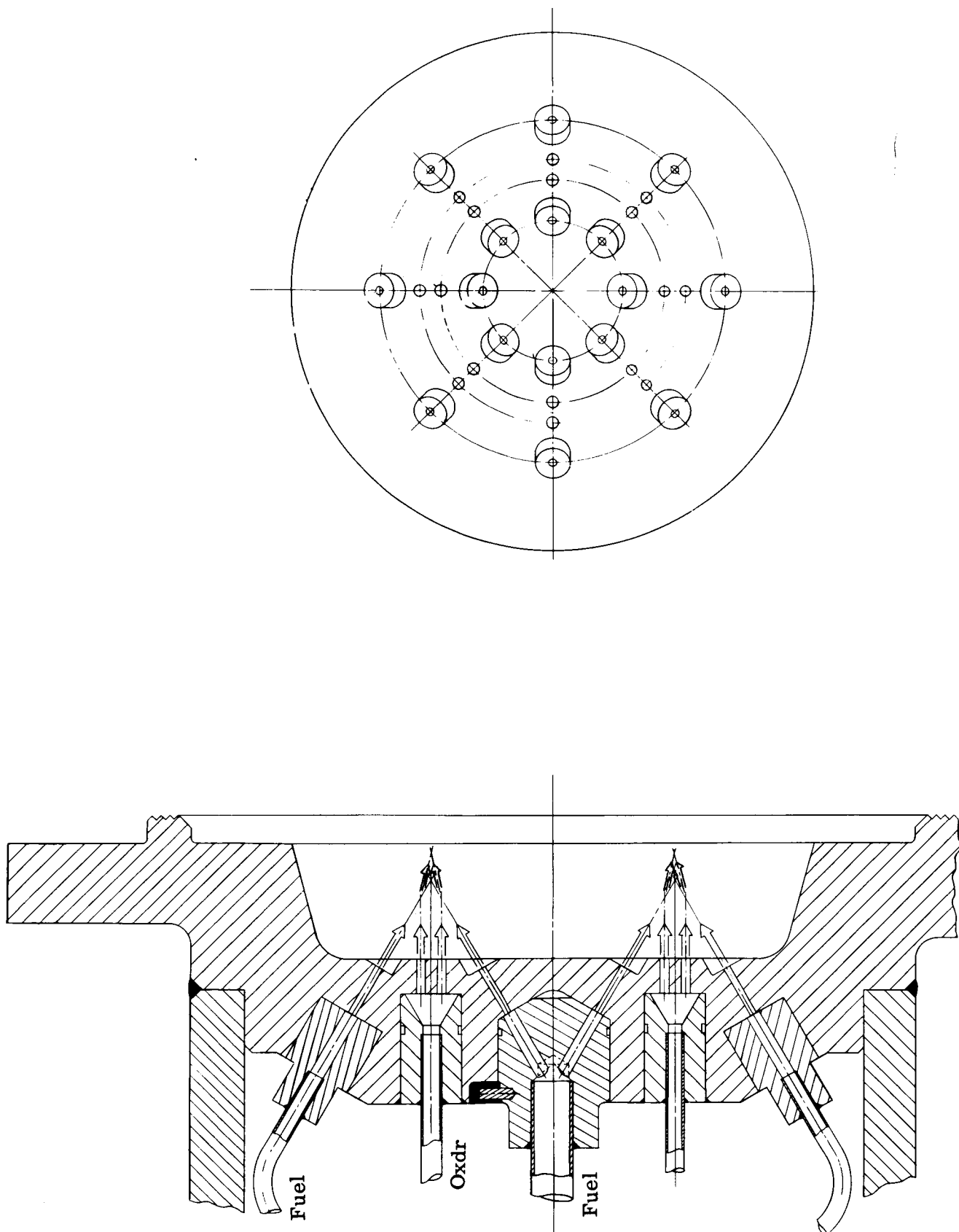


Figure V-70. 8 Element - Split Triplet Injector TS-4 (Test Configuration)

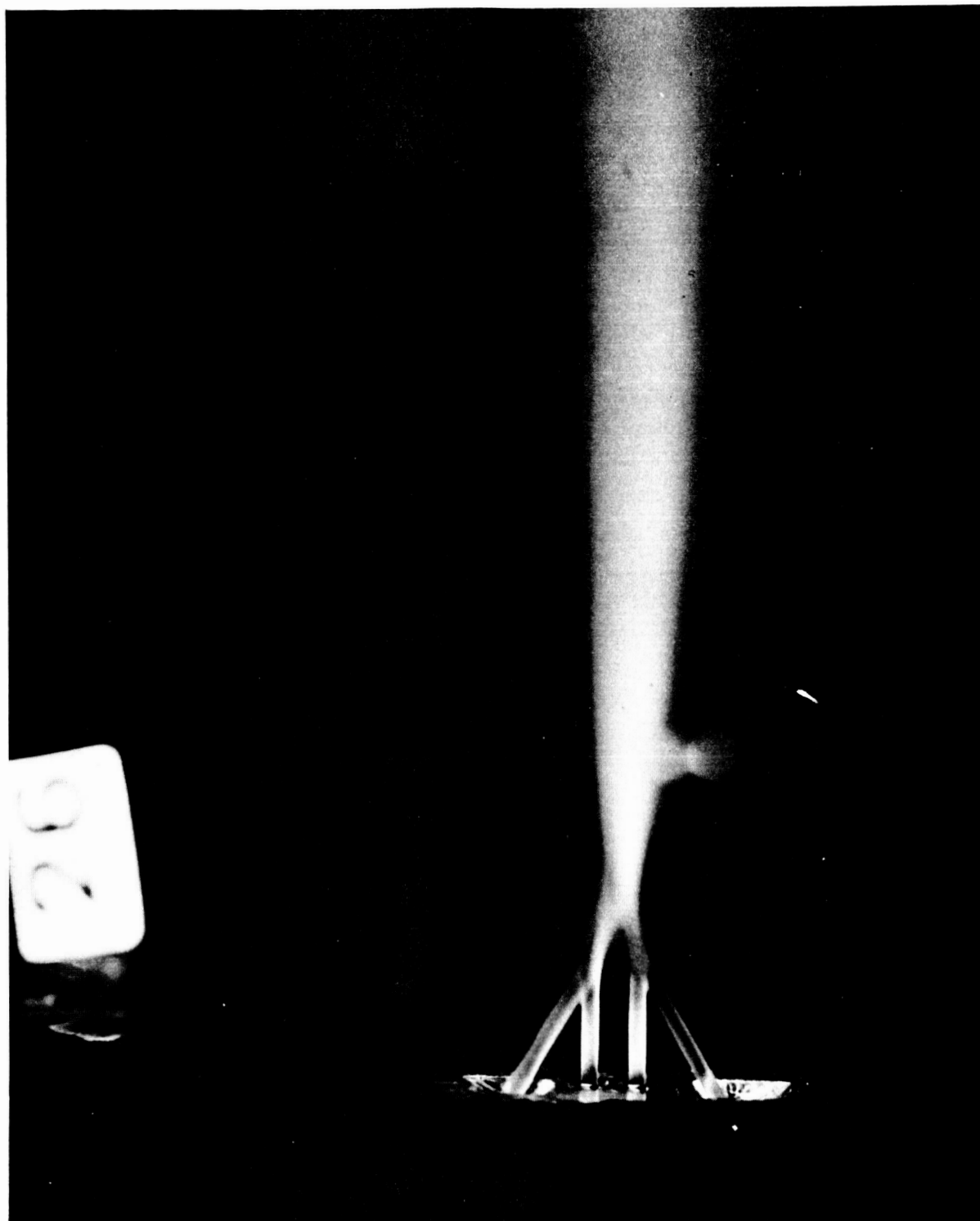


Figure V-71. H₂O Flow - Split Triplet

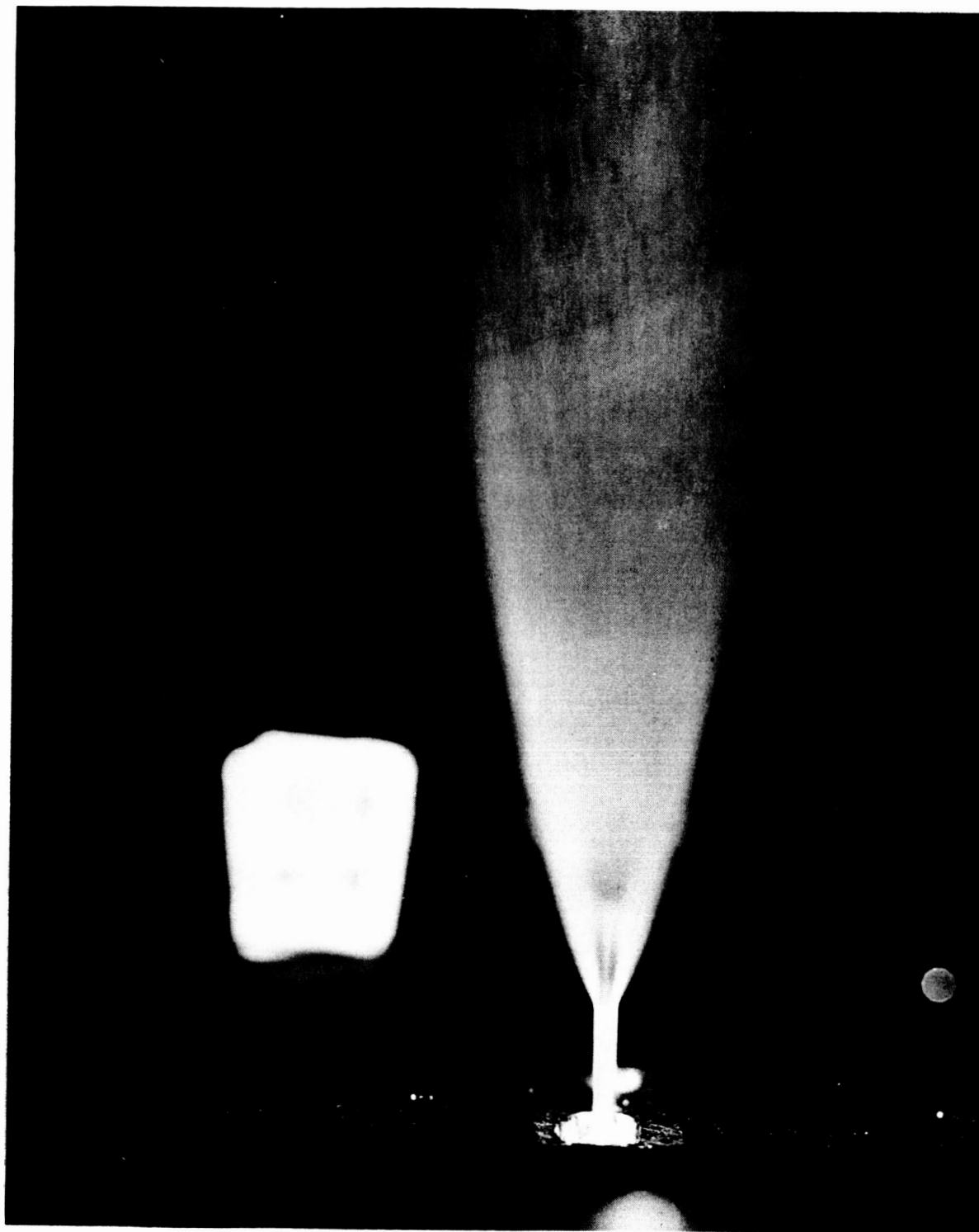
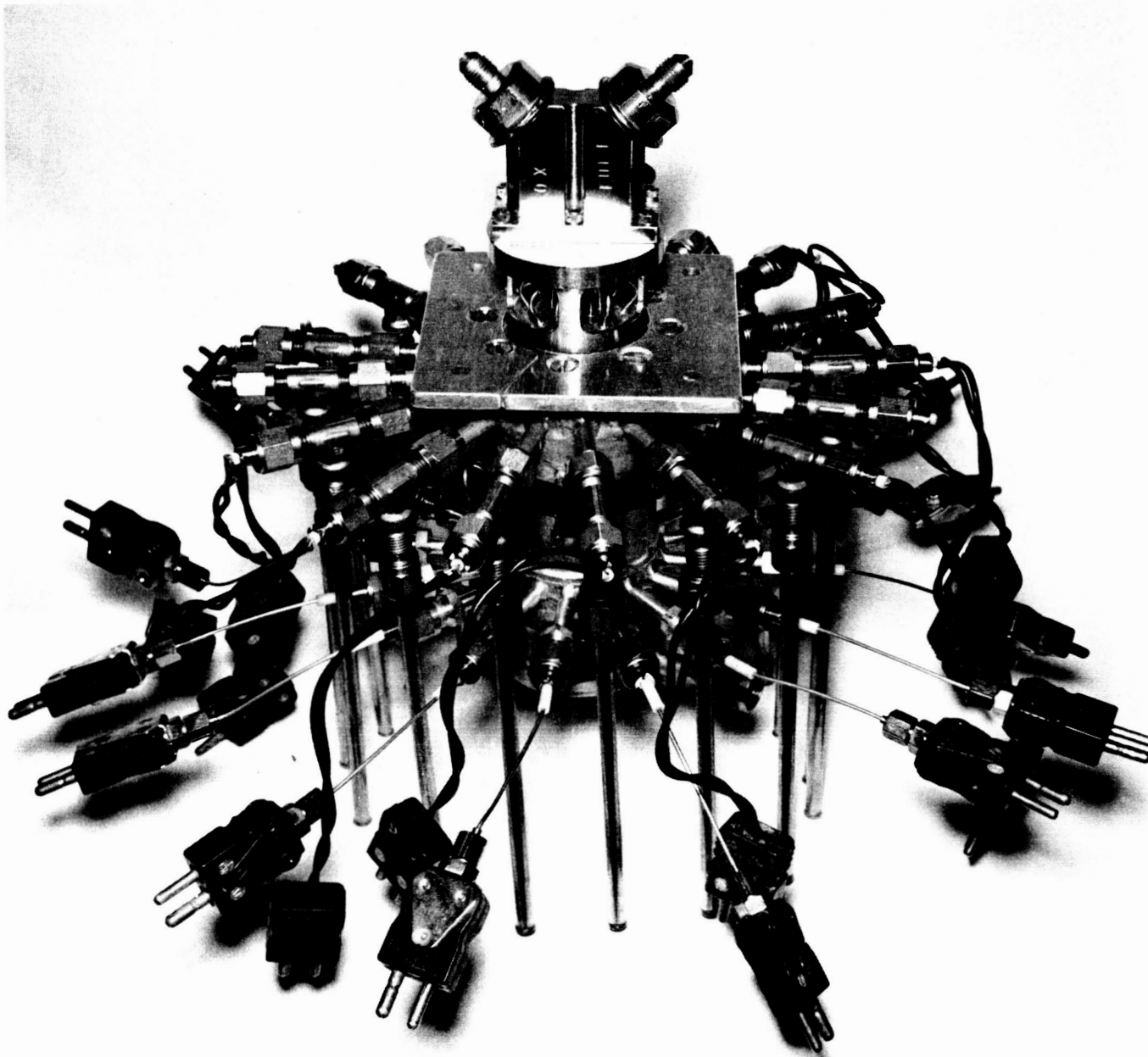


Figure V-72. H₂O Flow - Split Triplet



203382

Figure V-73. Typical Installation of Injector in Water Cooled Chamber

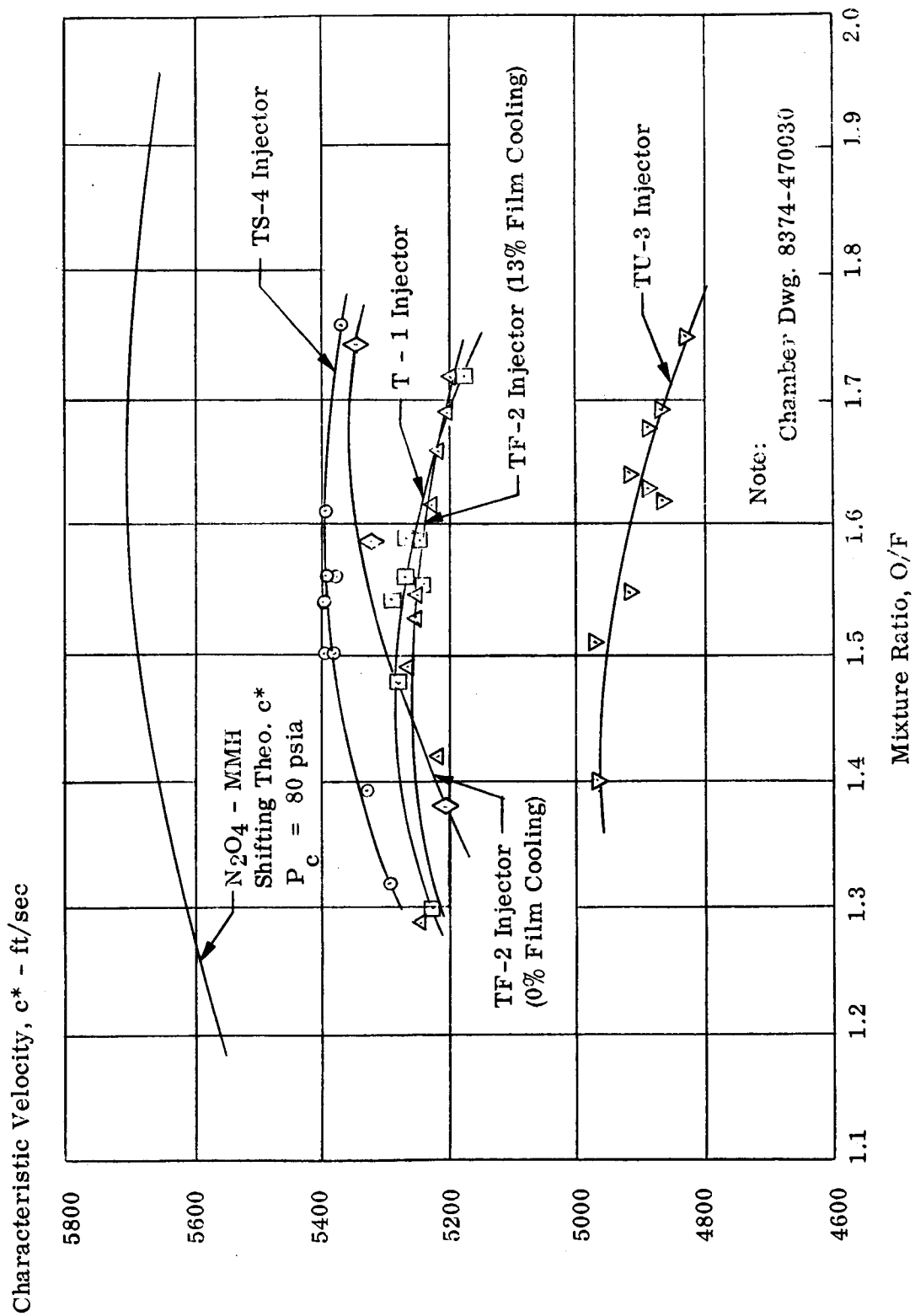


Figure V-74. Injector Performance in a Water Cooled Chamber

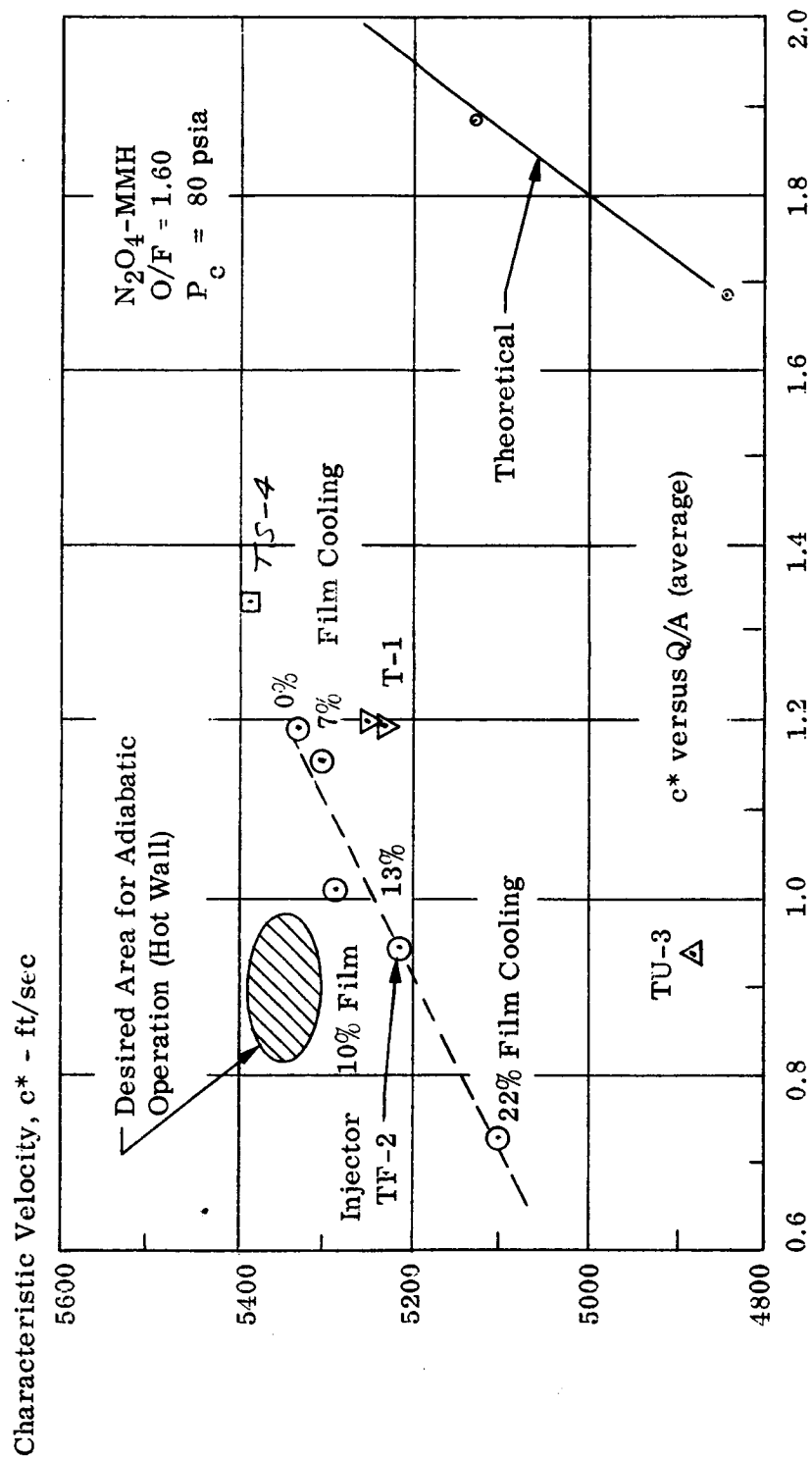


Figure V-75. Average Total Q/A - BTU/sec/in.²

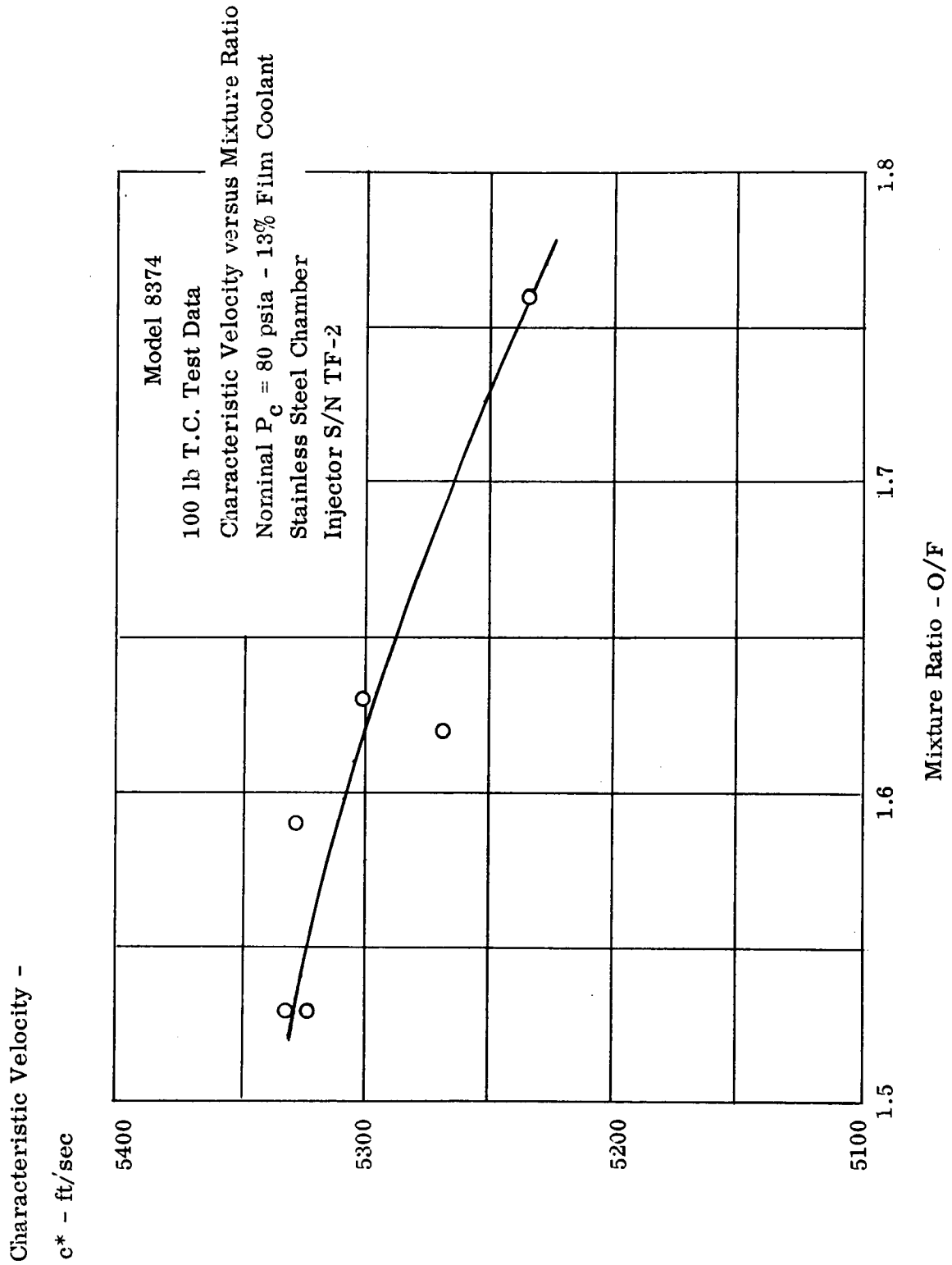


Figure V-76. 100 lb T.C. Test Data Characteristic Velocity versus Mixture Ratio

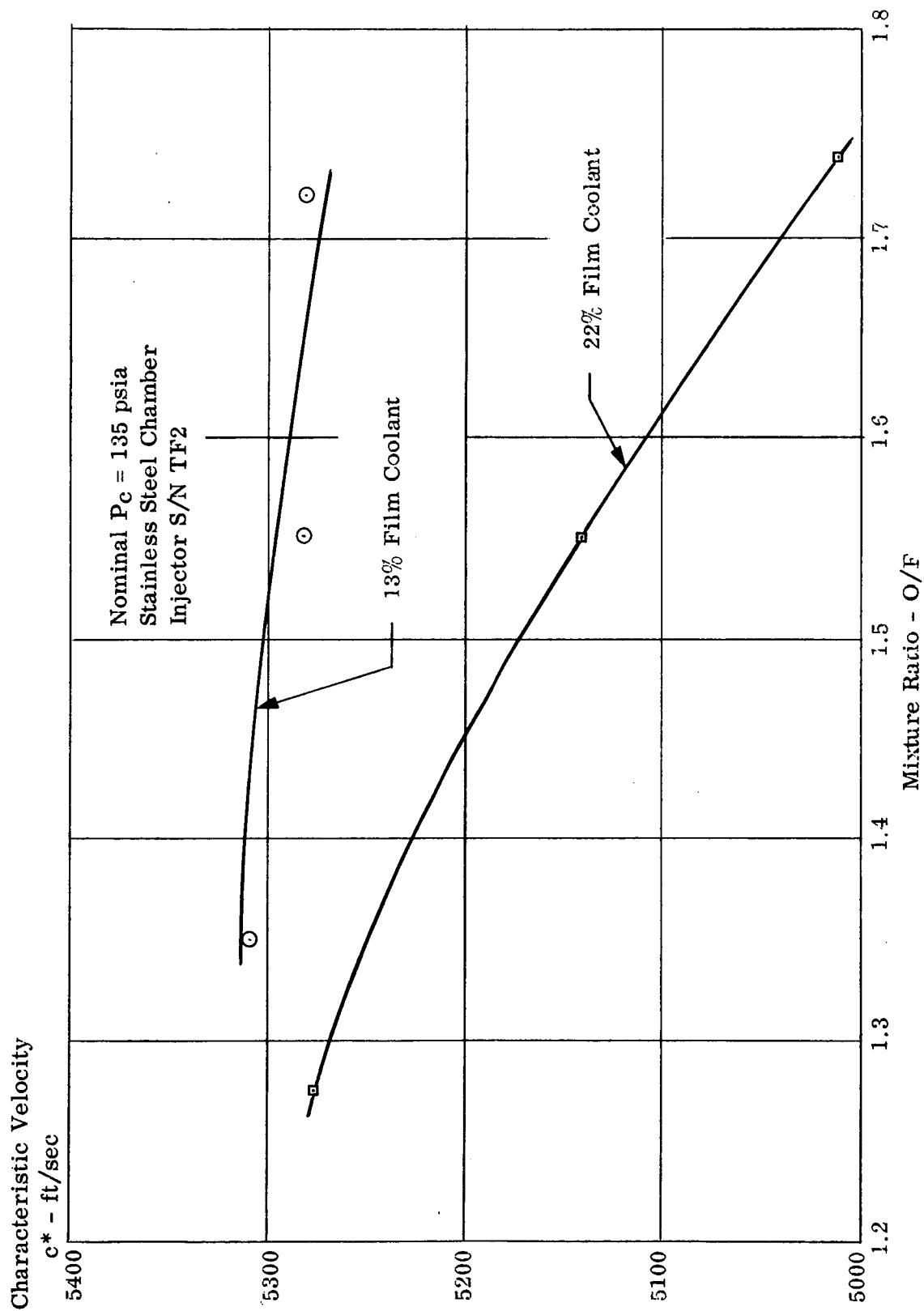


Figure V-77. 100 lb T.C. Test Data Characteristic Velocity versus Mixture Ratio

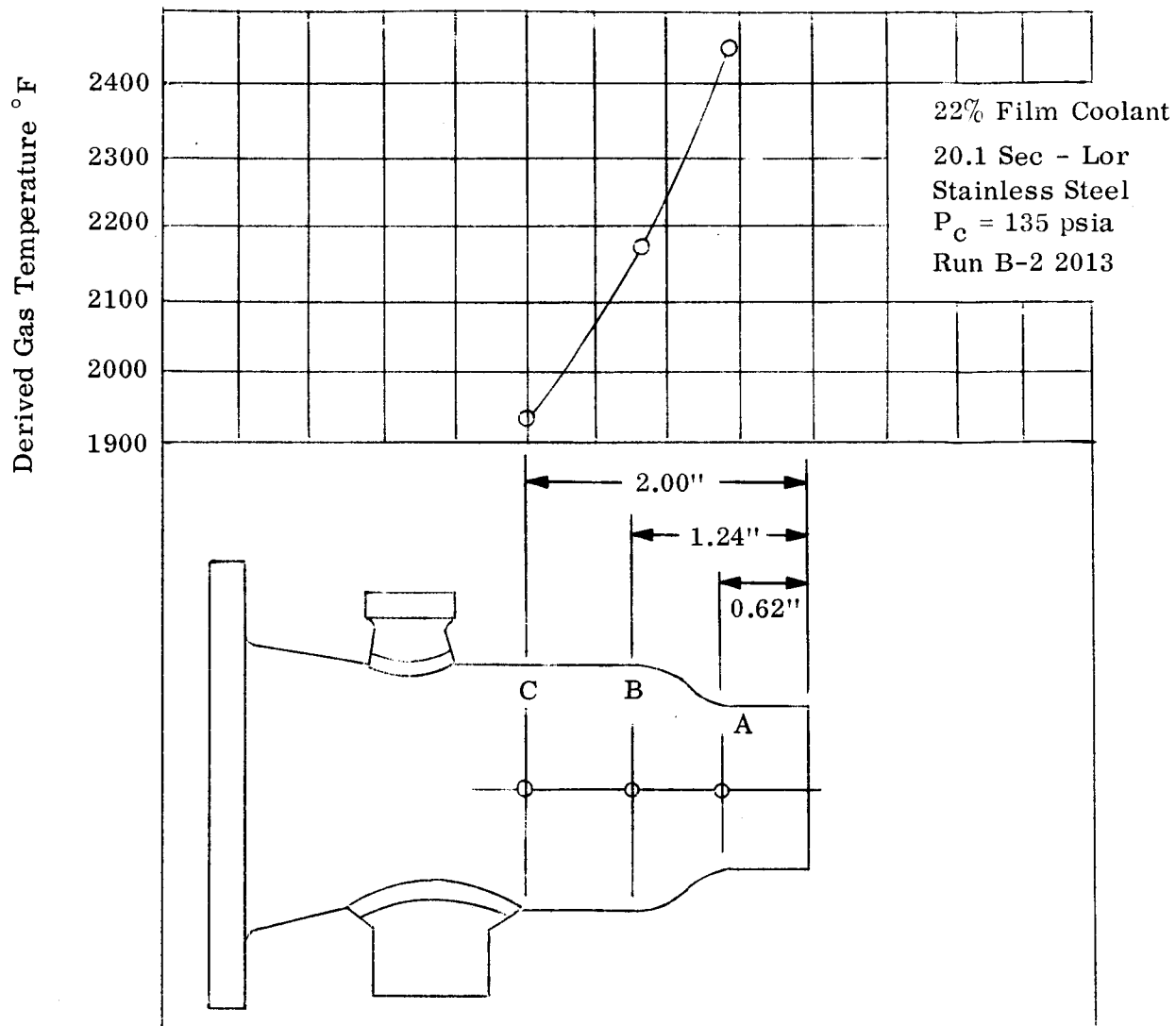


Figure V-78. Thermocouple Location

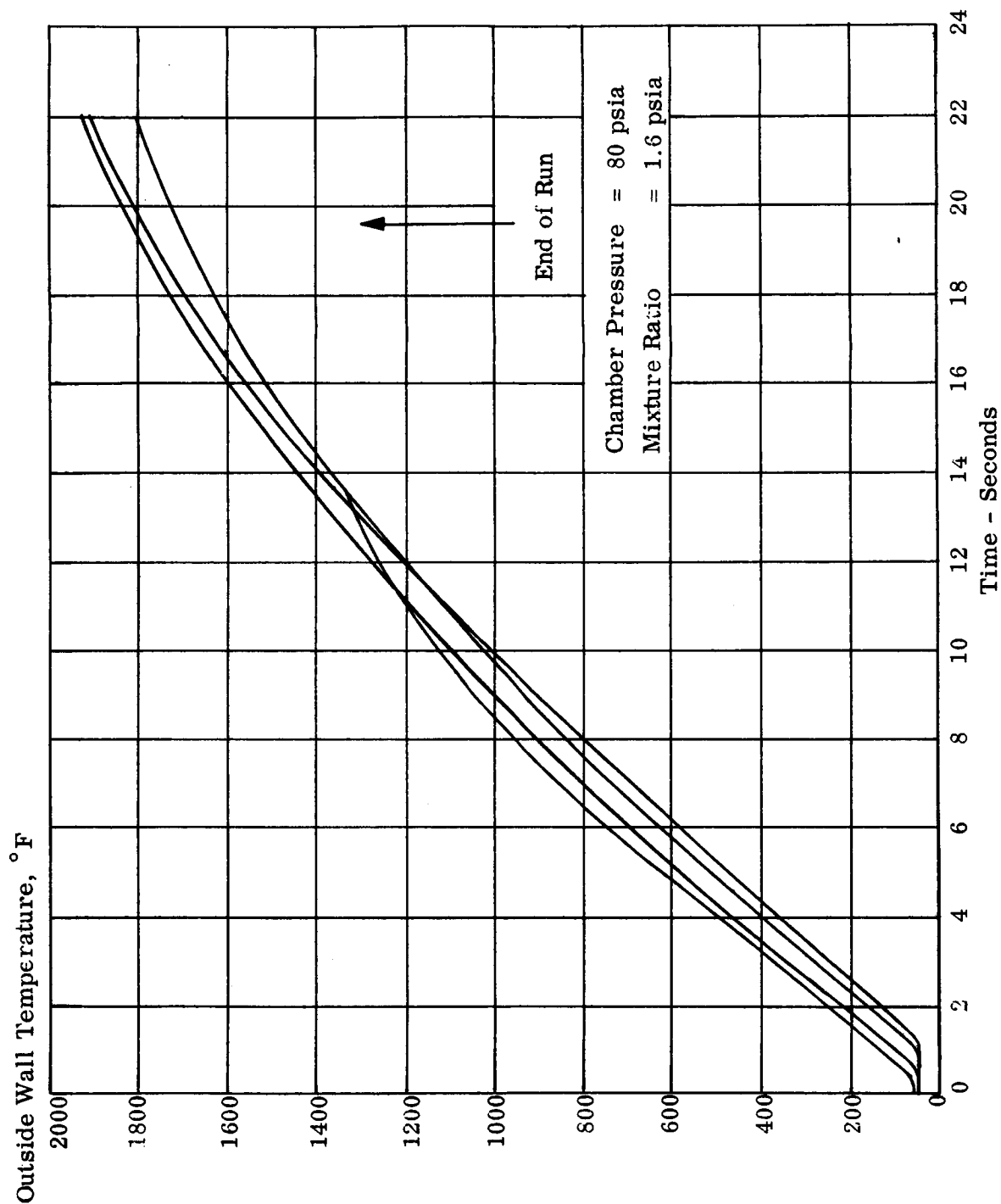


Figure V-79. Wall Temperature versus Time - Throat Station Test 2B-1985.
Stainless Steel Chamber, Film Cooled Injector S/N TF-2

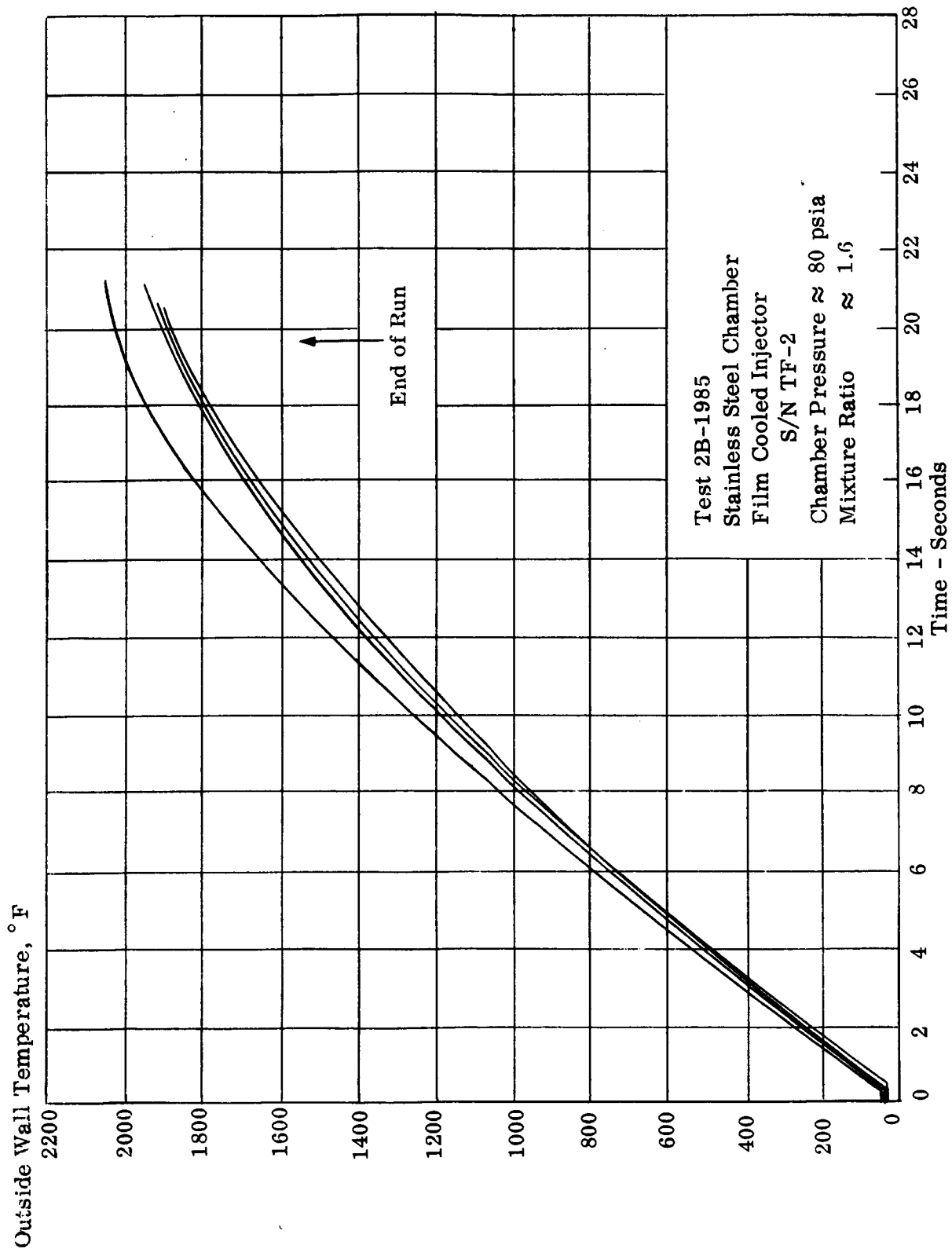


Figure V-80. Wall Temperature versus Time - Chamber Near Convergent Nozzle

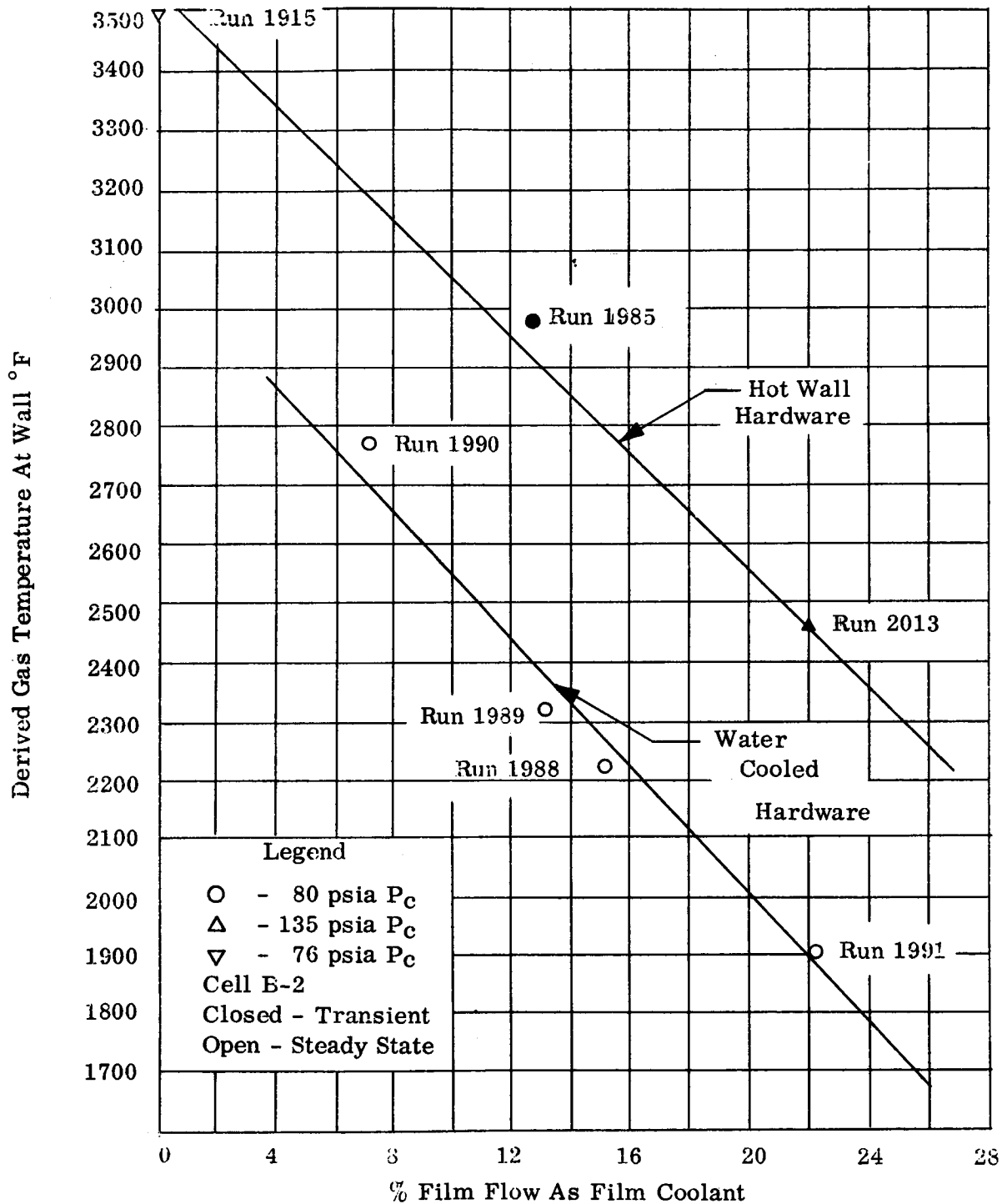


Figure V-81. Wall Gas Temperature versus Percent Film Coolant

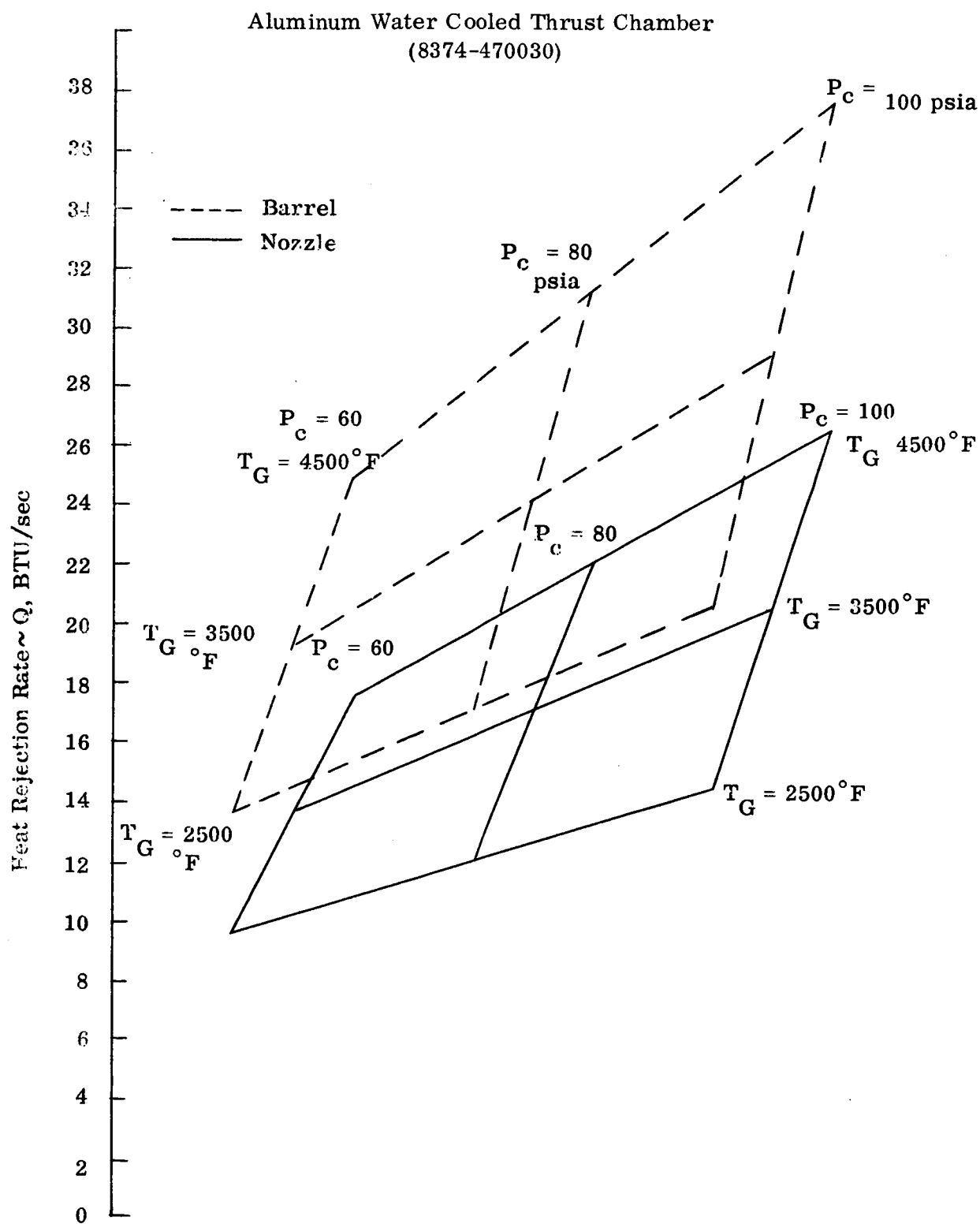


Figure V-82. Heat Rejection Rate versus Driving Temperature

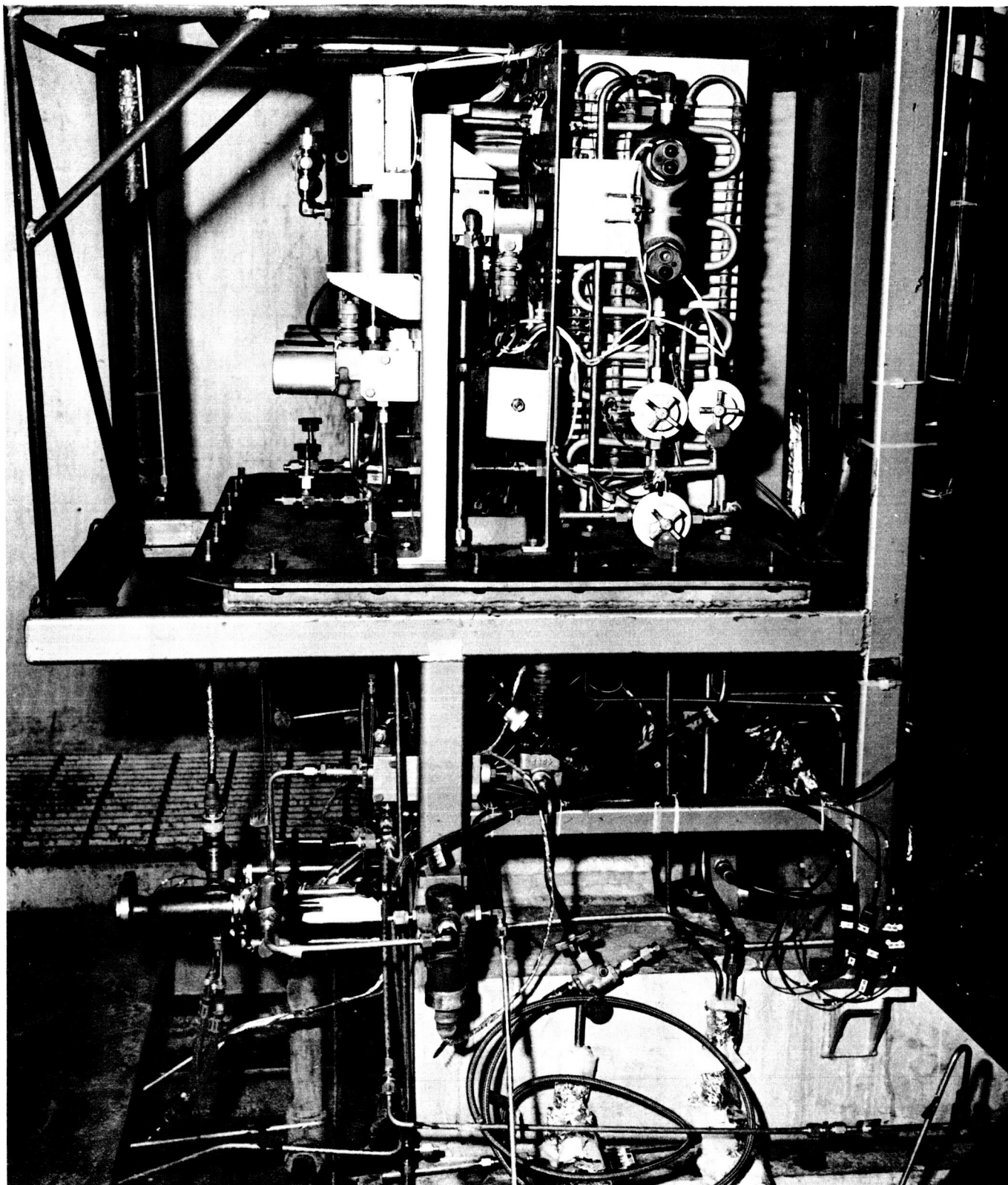


Figure V-83. Pulse Mode Stand - W/O Cover

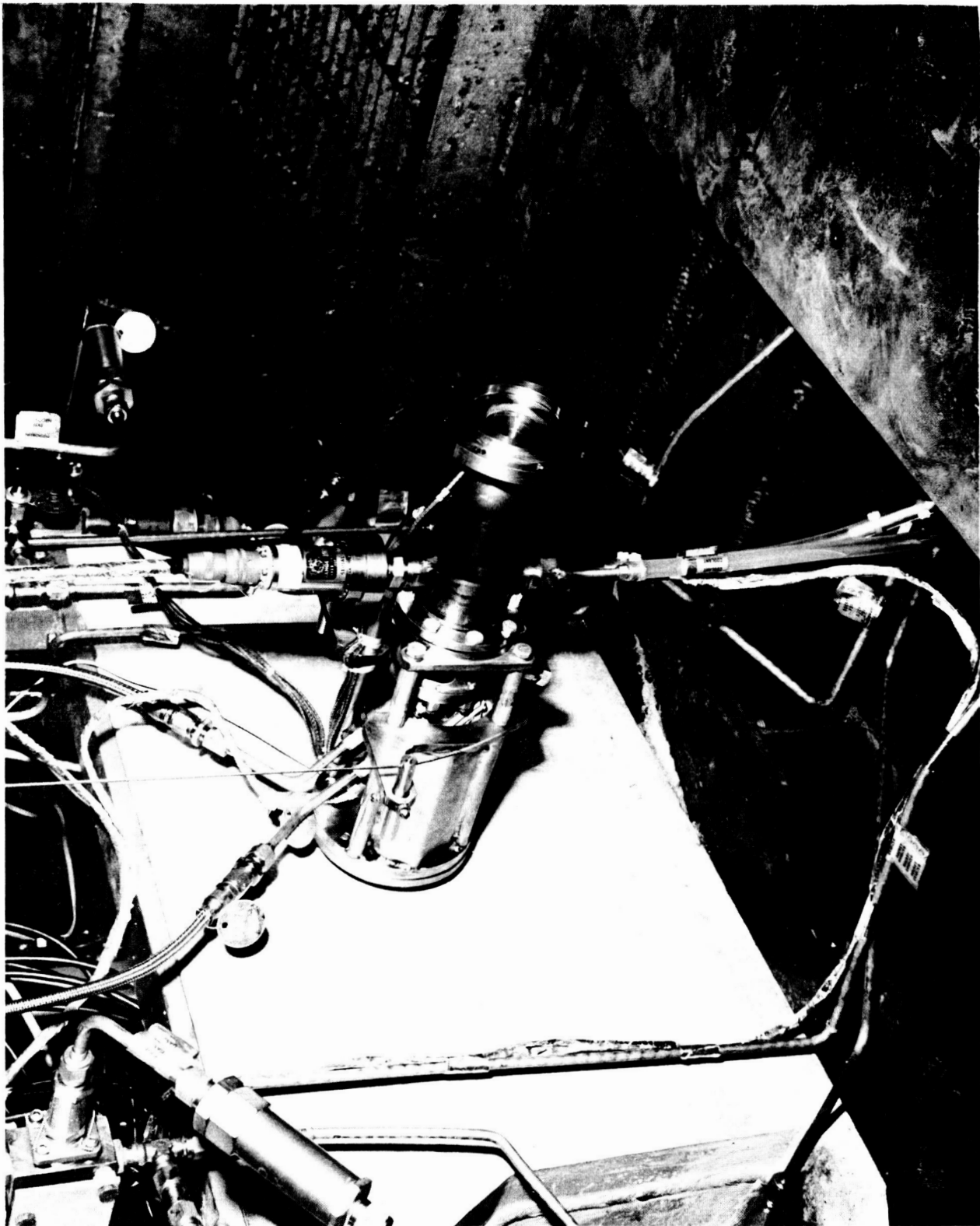
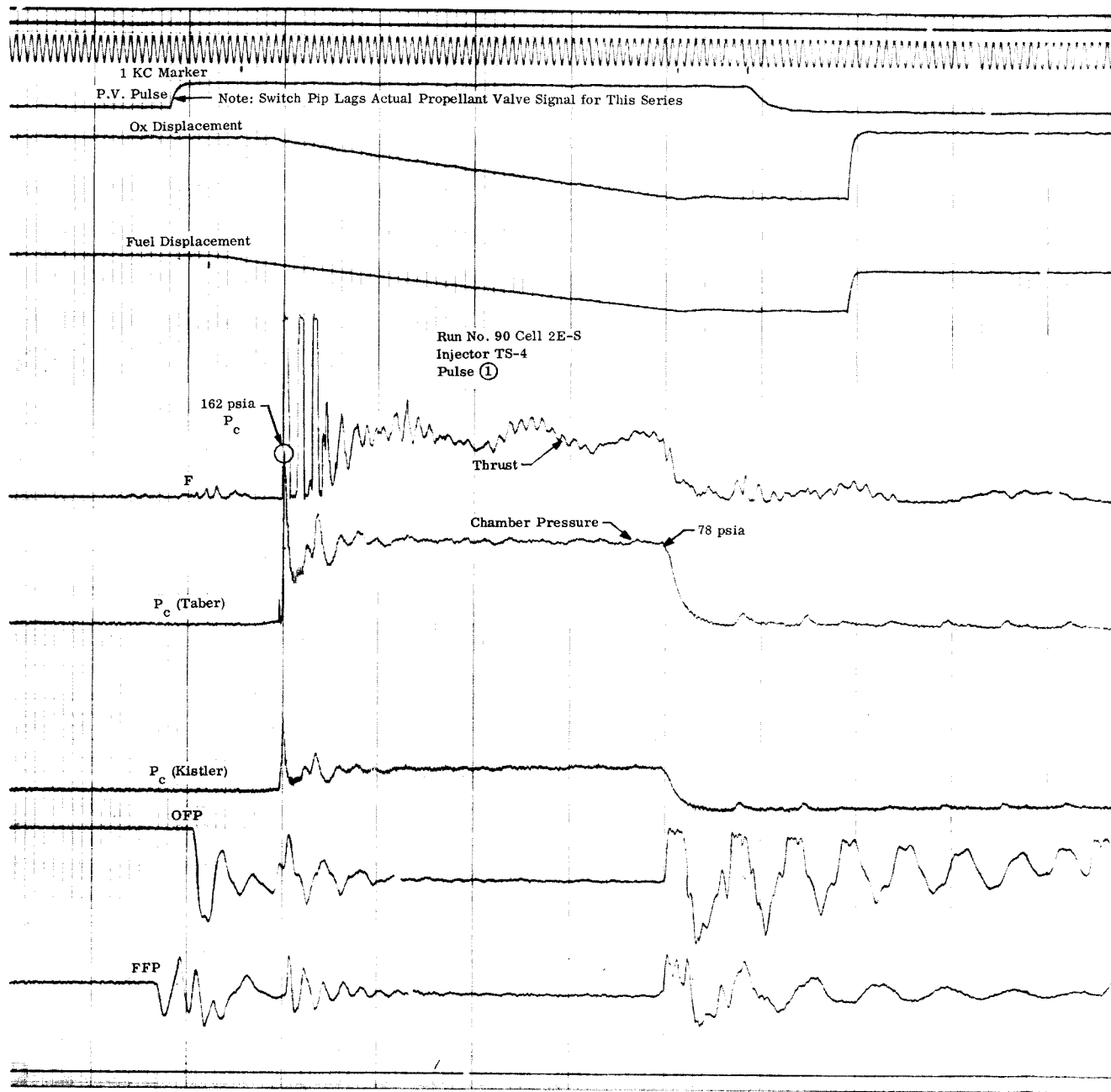
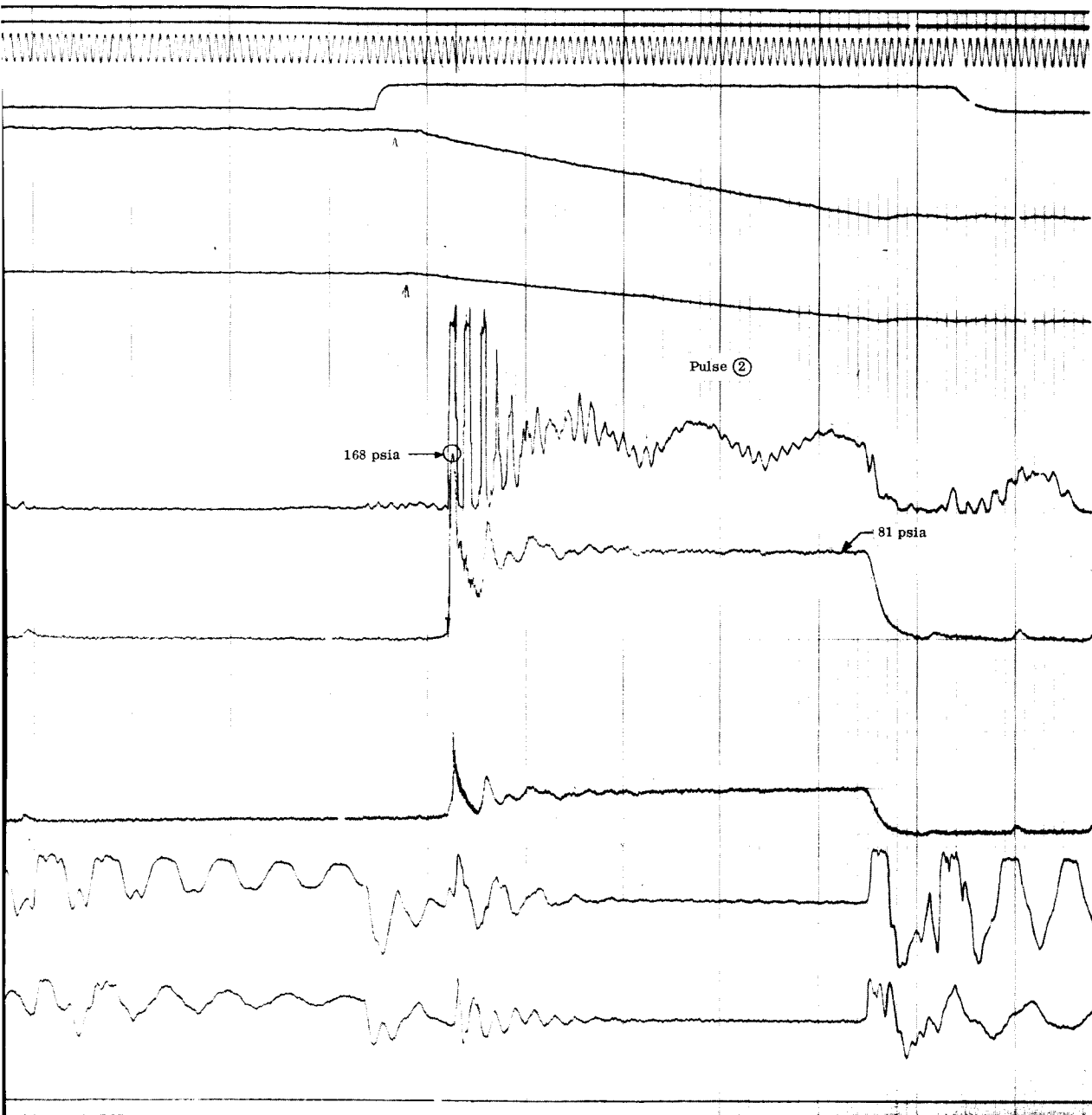


Figure V-84. Pulse Mode Stand - Chamber Closeup





R29B

Figure V-85. Typical Oscillograph Trace

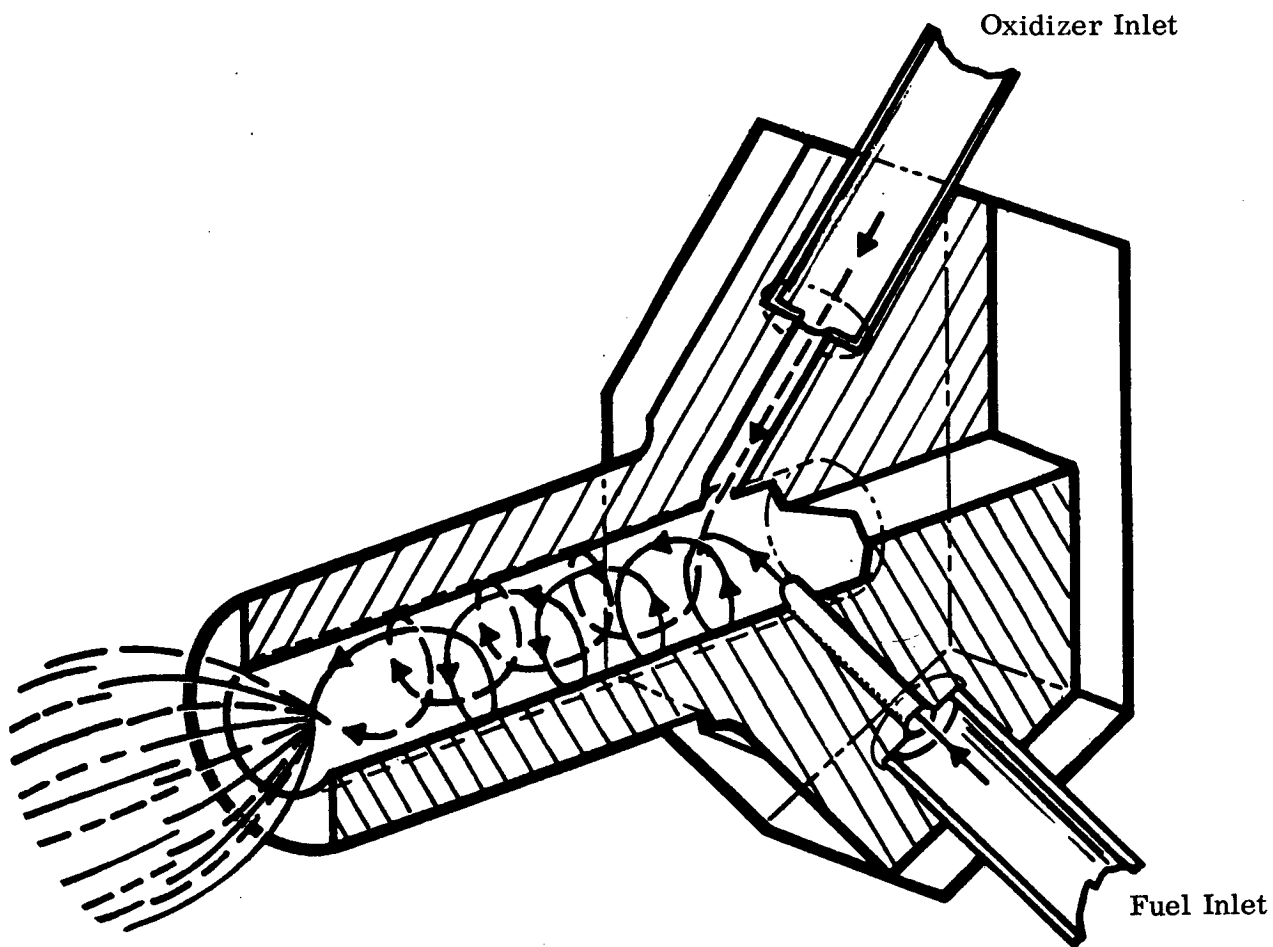


Figure V-86. Preigniter Model

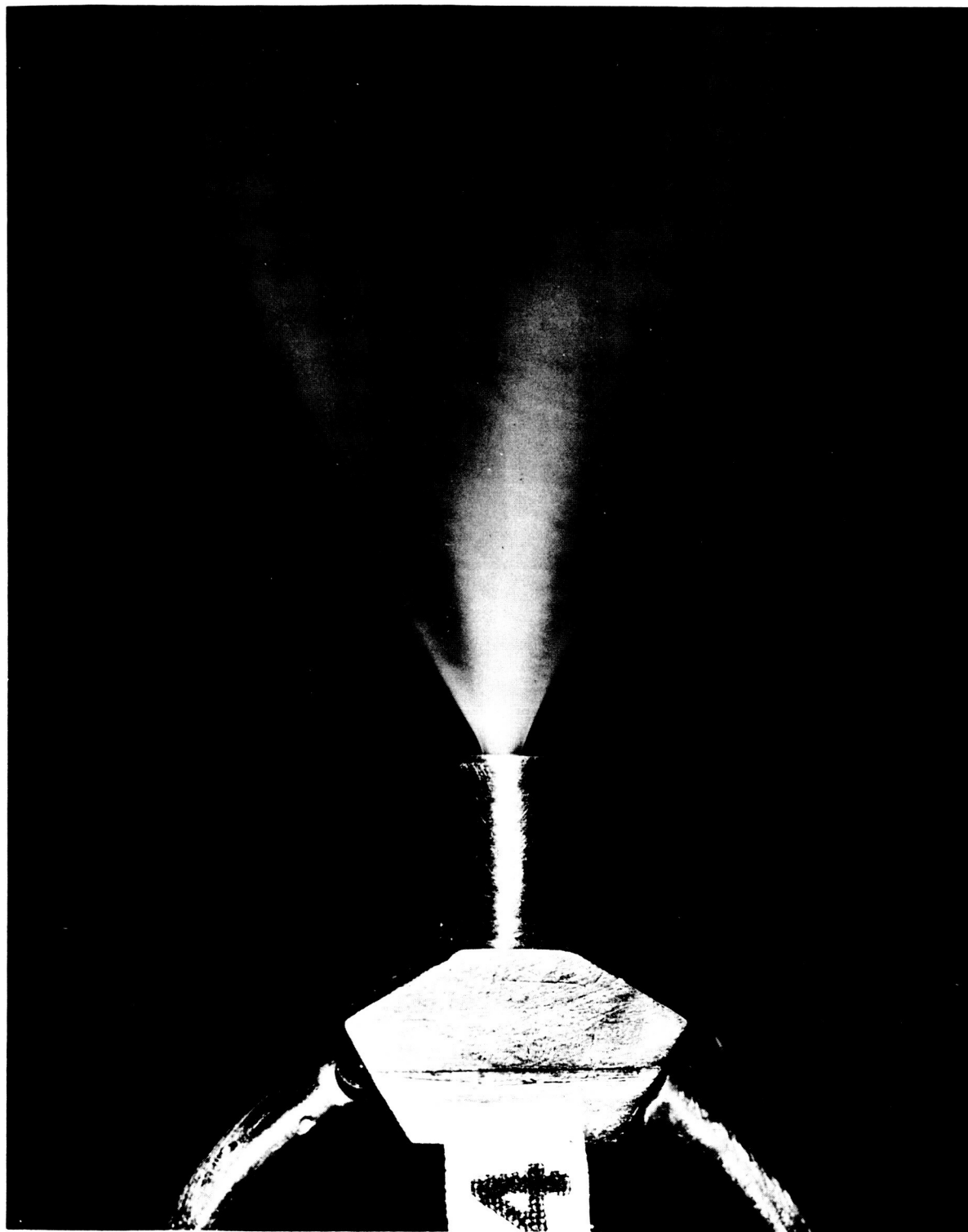


Figure V-87. H_2O Flow Preigniter

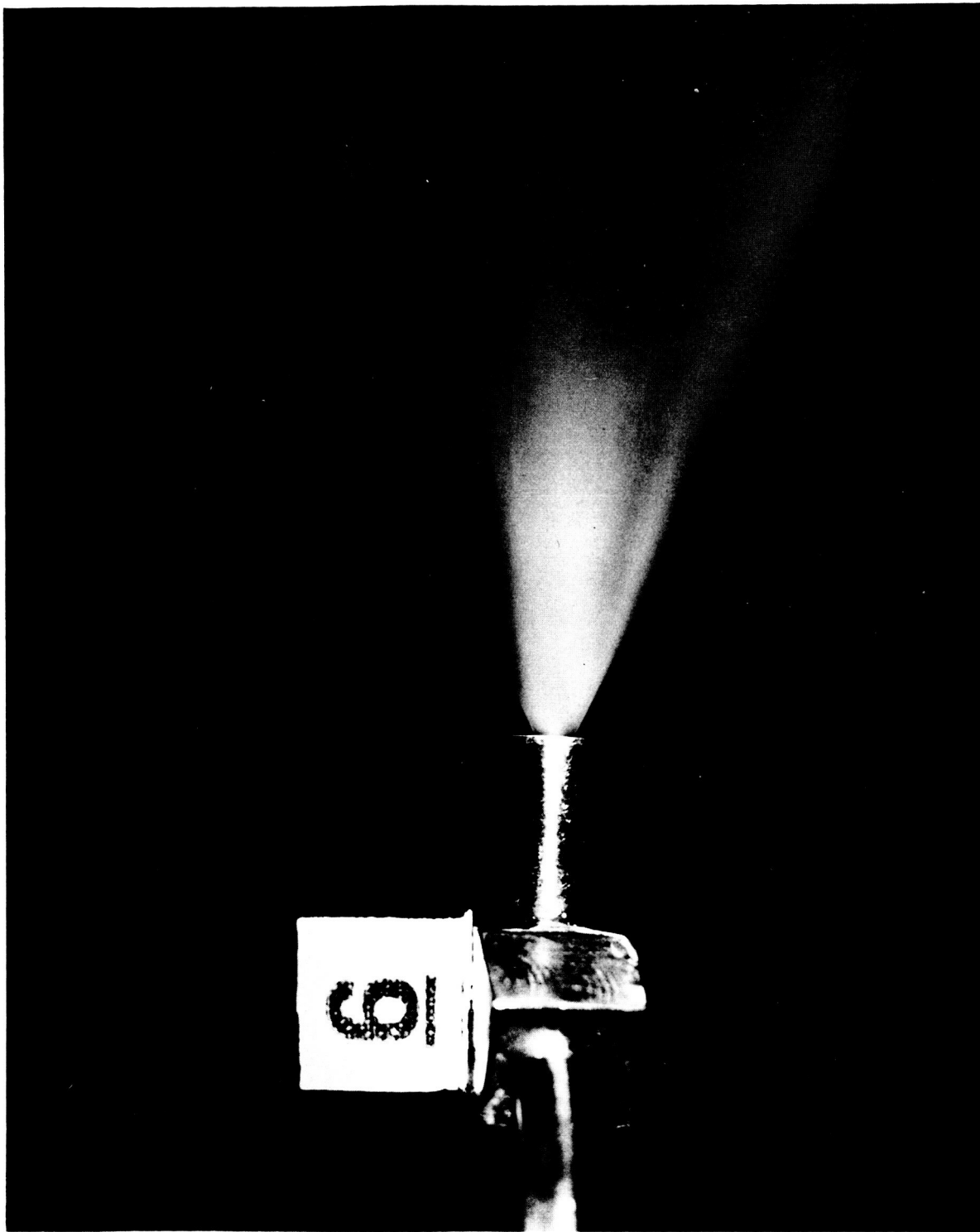


Figure V-88. H₂O Flow Preigniter

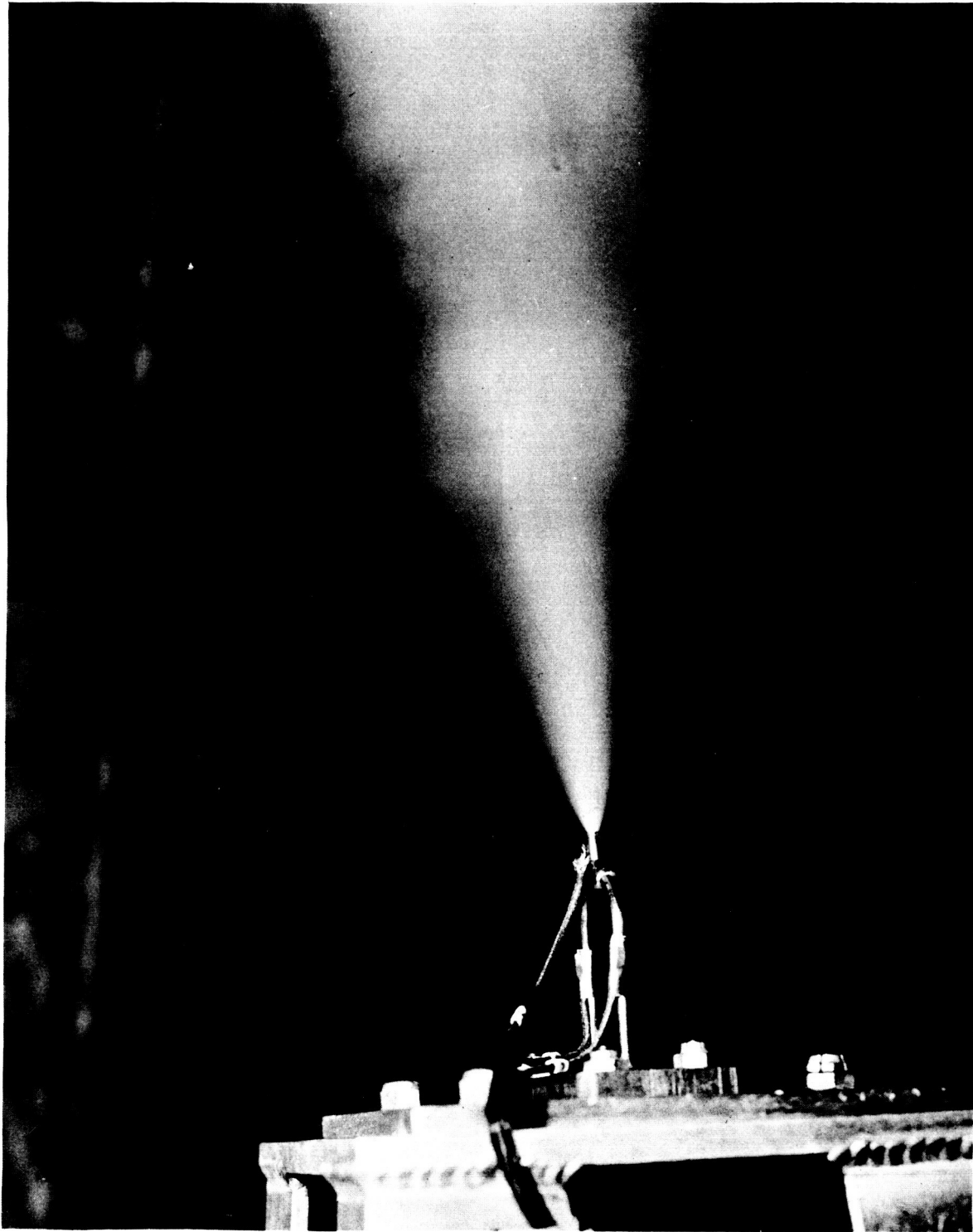


Figure V-89. Preigniter Hot Fire Test

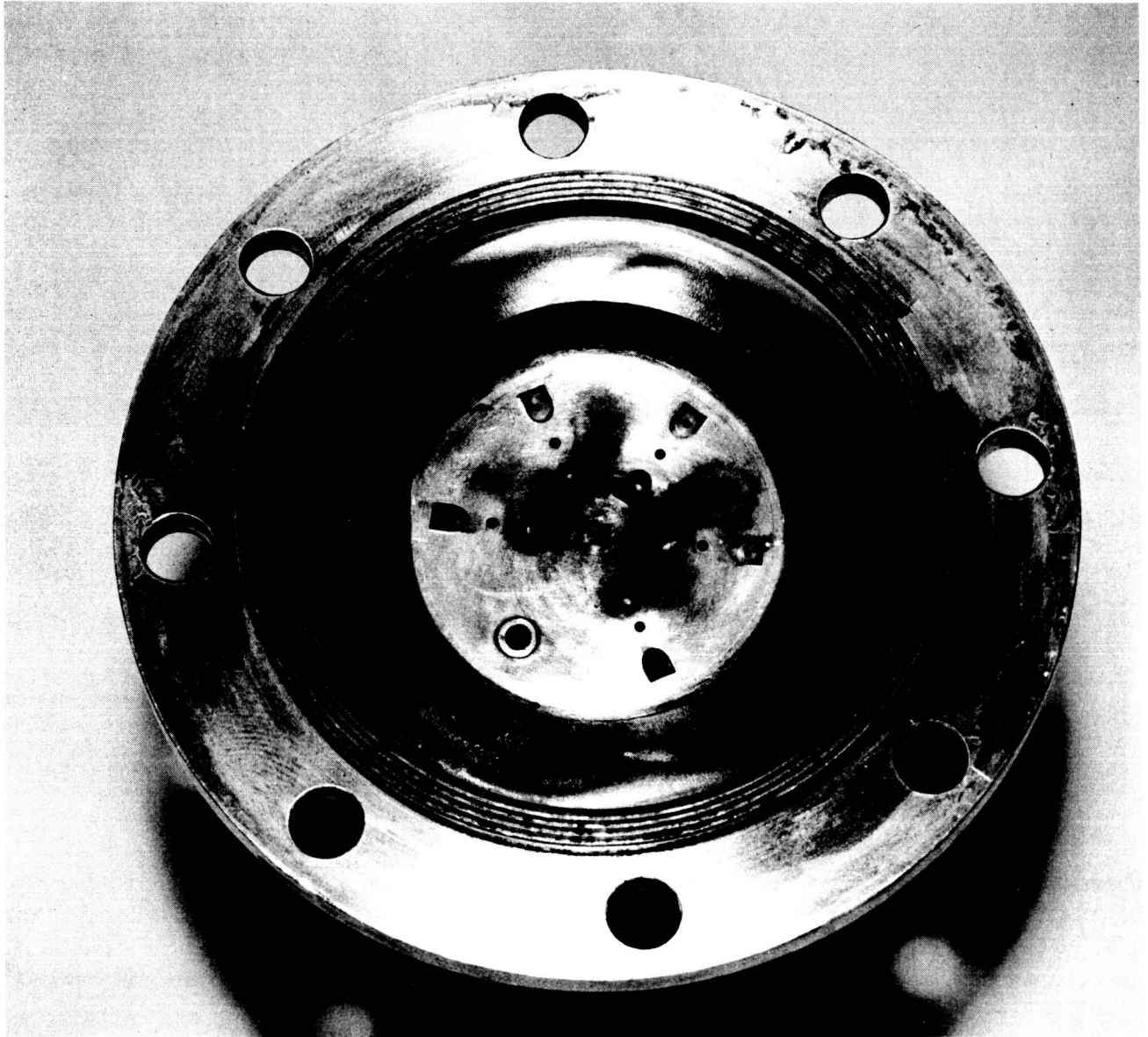
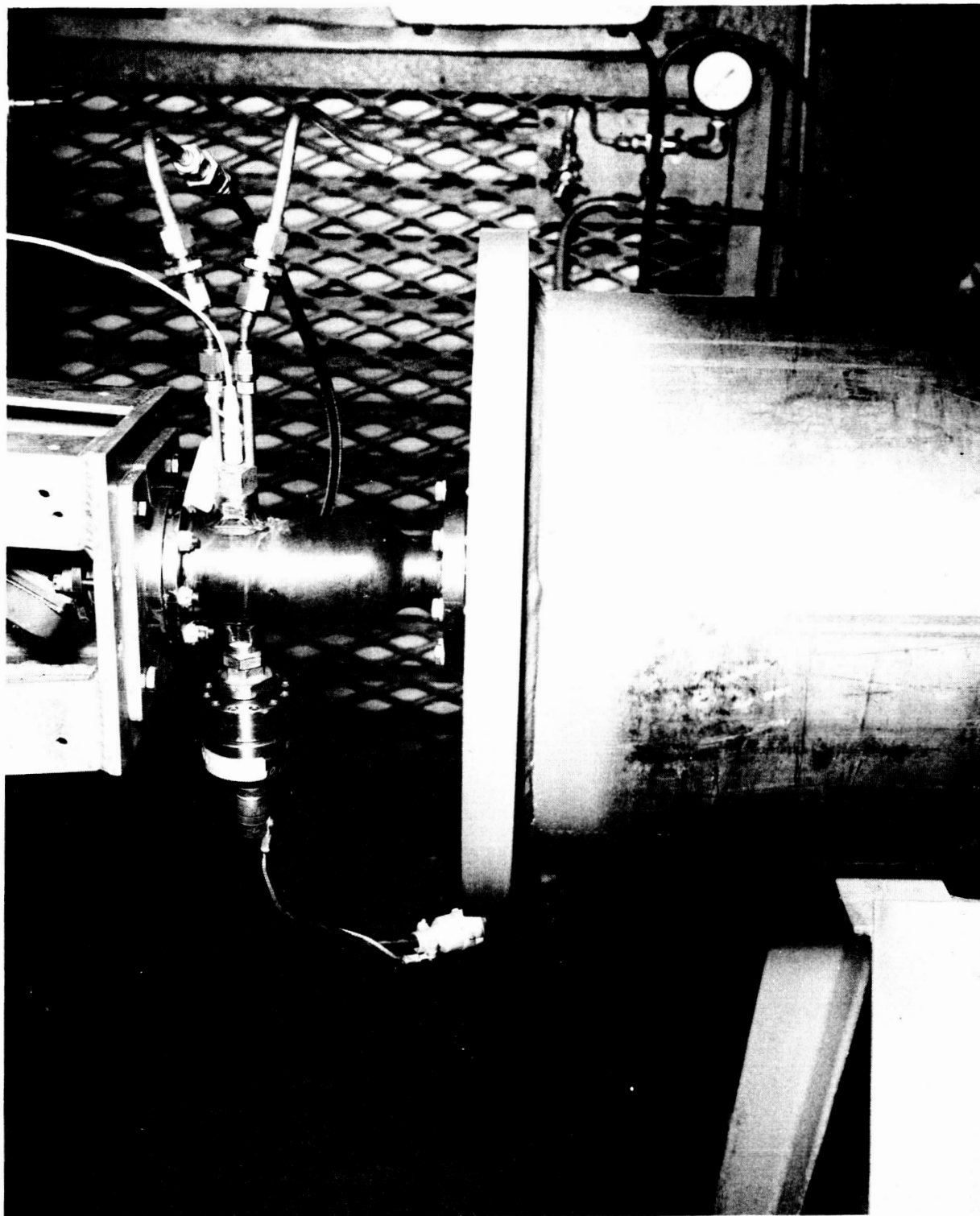


Figure V-90. Preigniter Injector

203384



202146

Figure V-91. Preigniter Injector Test Setup

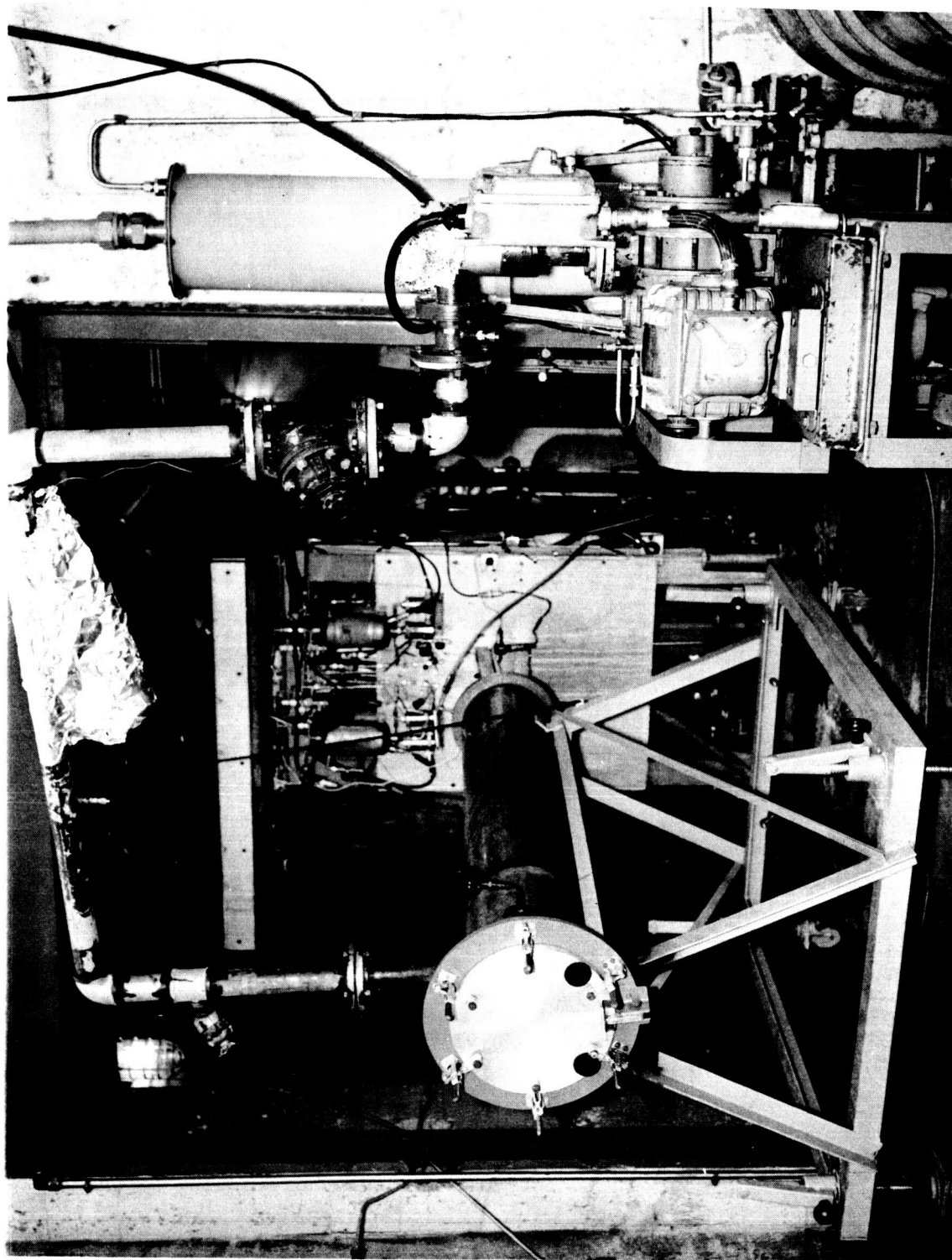
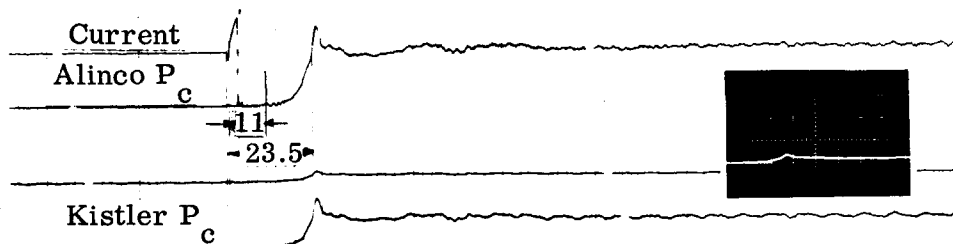
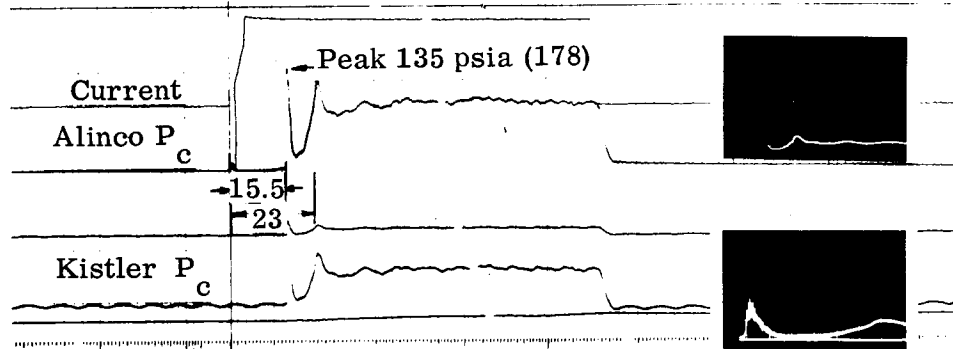


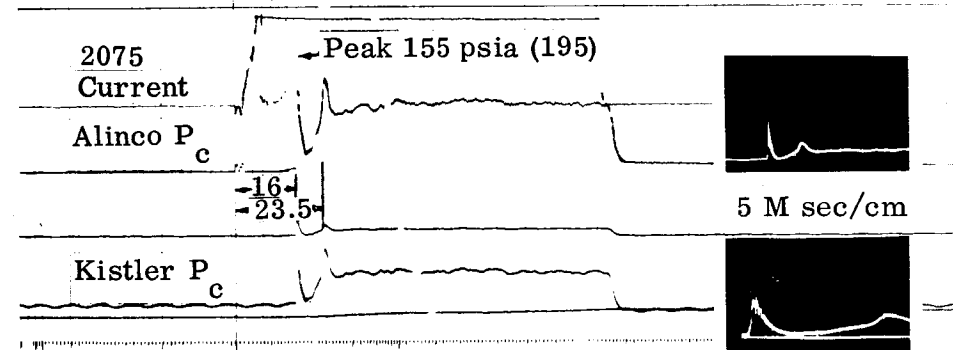
Figure V-92. Vacuum Tank Setup



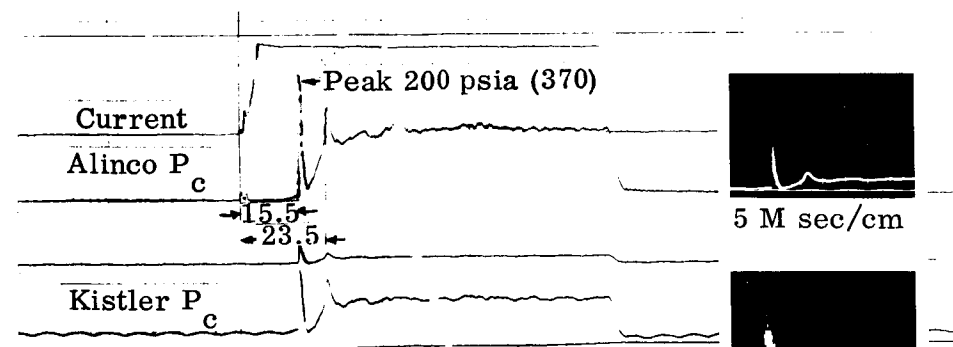
Run 2072 Sea Level Prop. Temp +60° F



Run 2074 160,000 ft (700 μ) Prop. Temp +60°



Run 2075 160,000 ft (700 μ) Prop. Temp +60° 1.25 M sec/cm



Run 2076 180,000 ft (450 μ) Prop. Temp +60° 1.25 M sec/cm

Figure V-93. Without Preigniter

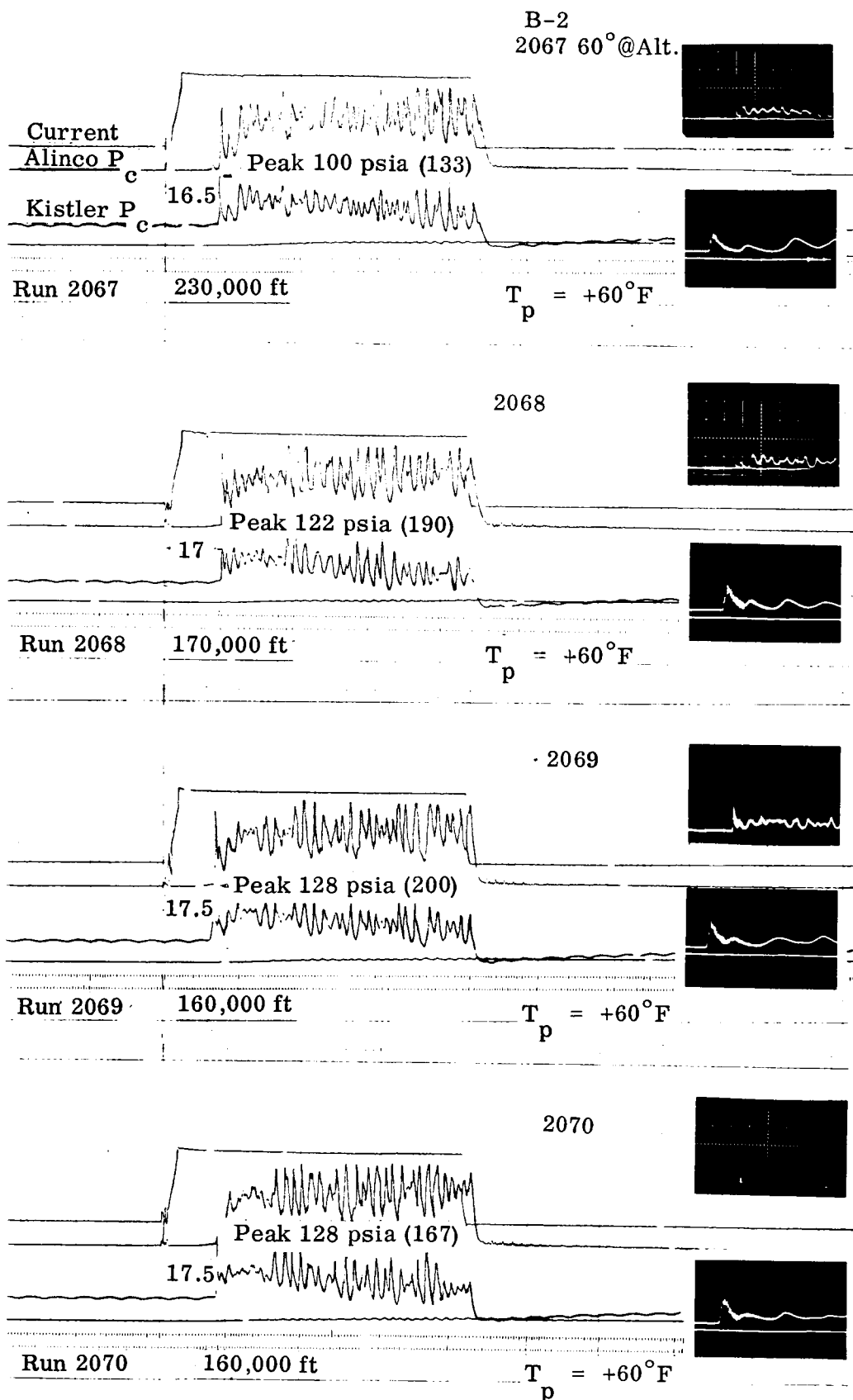
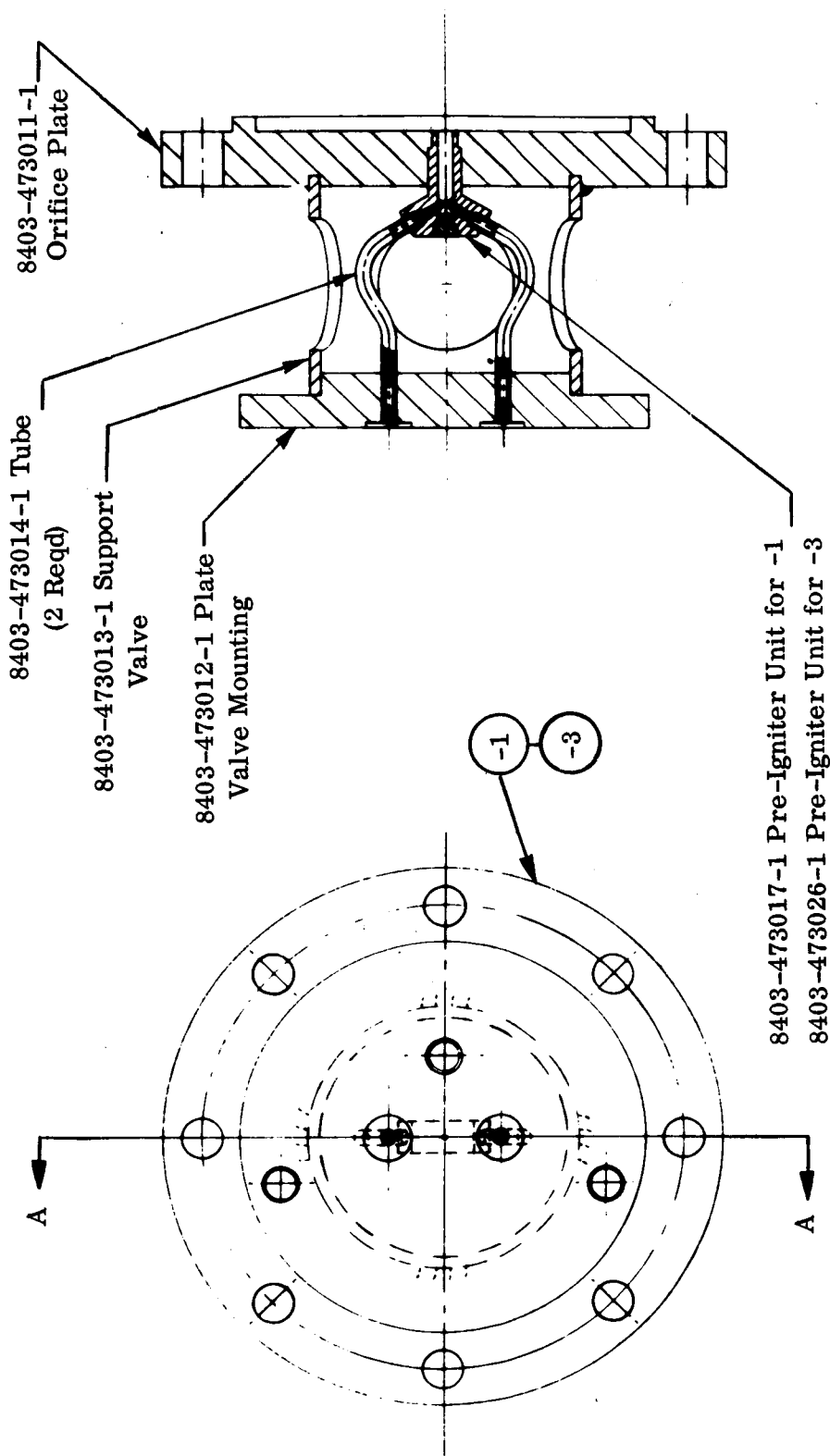
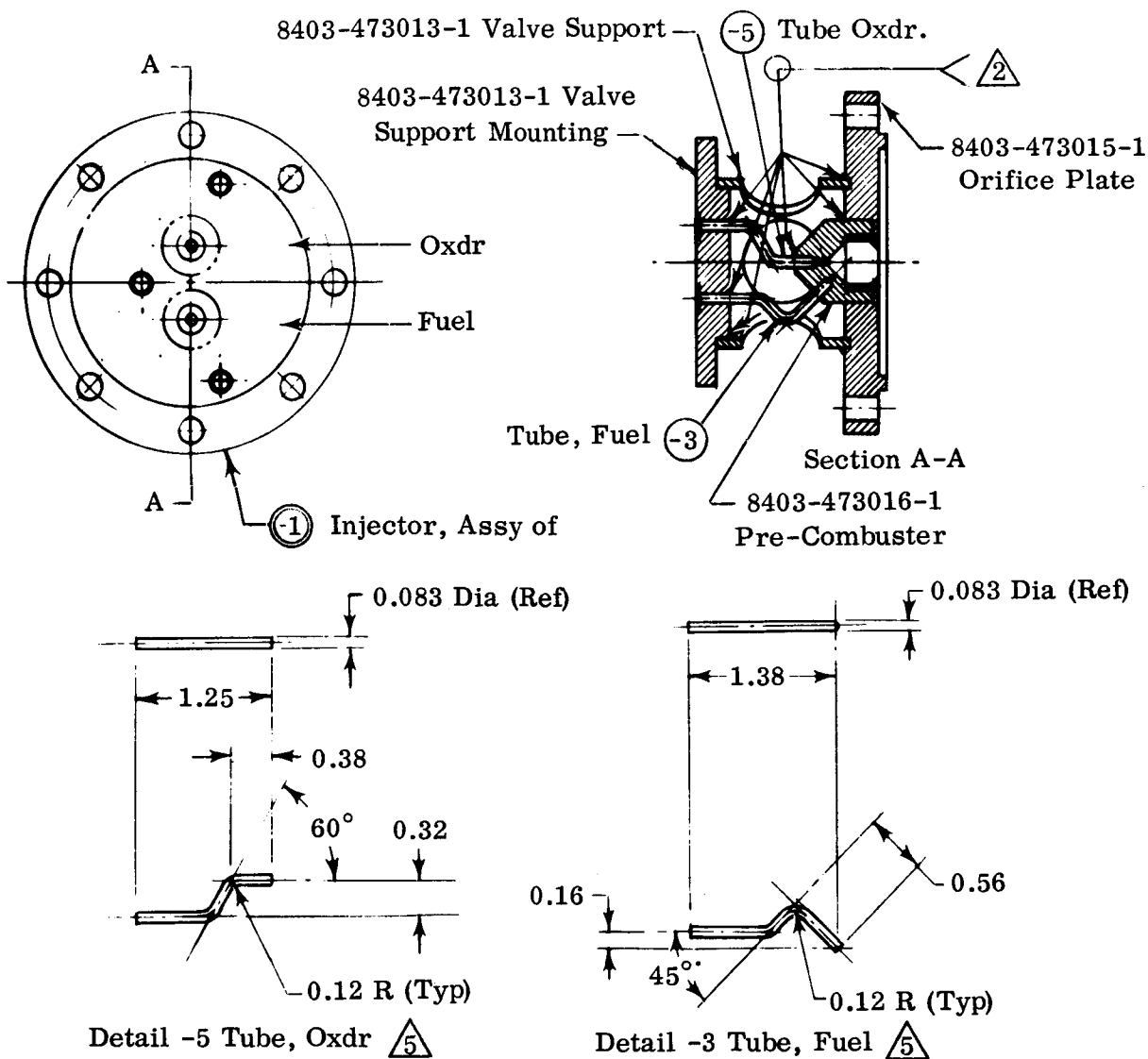


Figure V-94. With Preigniter



- 4 Provide protective covering for sealing surfaces during shipping, handling & storage.
- 3 Clean all details per BPS 4138 prior to assembly.
- 2 Following brazing operation, thoroughly clean & remove any trace of Stop-Off per instructions of process and development lab.
- 1 All parts of this assembly are to be braze assembled. All brazing to be done simultaneously in process and development lab.

Figure V-95. Preigniter Element



- △5 Tube ends to be square and free of burrs and chips.
- △4 Tubing to be "Hypoflex" SST Tubing, AISI Type 304. Purchase from Superior Tube Company; Collegeville, Pa.
- △3 Provide protective covering for sealing surfaces during shipping, handling and storage.
- △2 Brazing to be accomplished in process and development lab following welding of 8403-473013-1 and 8403-473015-1. Following brazing thoroughly clean and remove all traces of Stop-Off.
- △1 Clean all parts per BPS 4138 prior to brazing.

Figure V-96. Precombustor Element

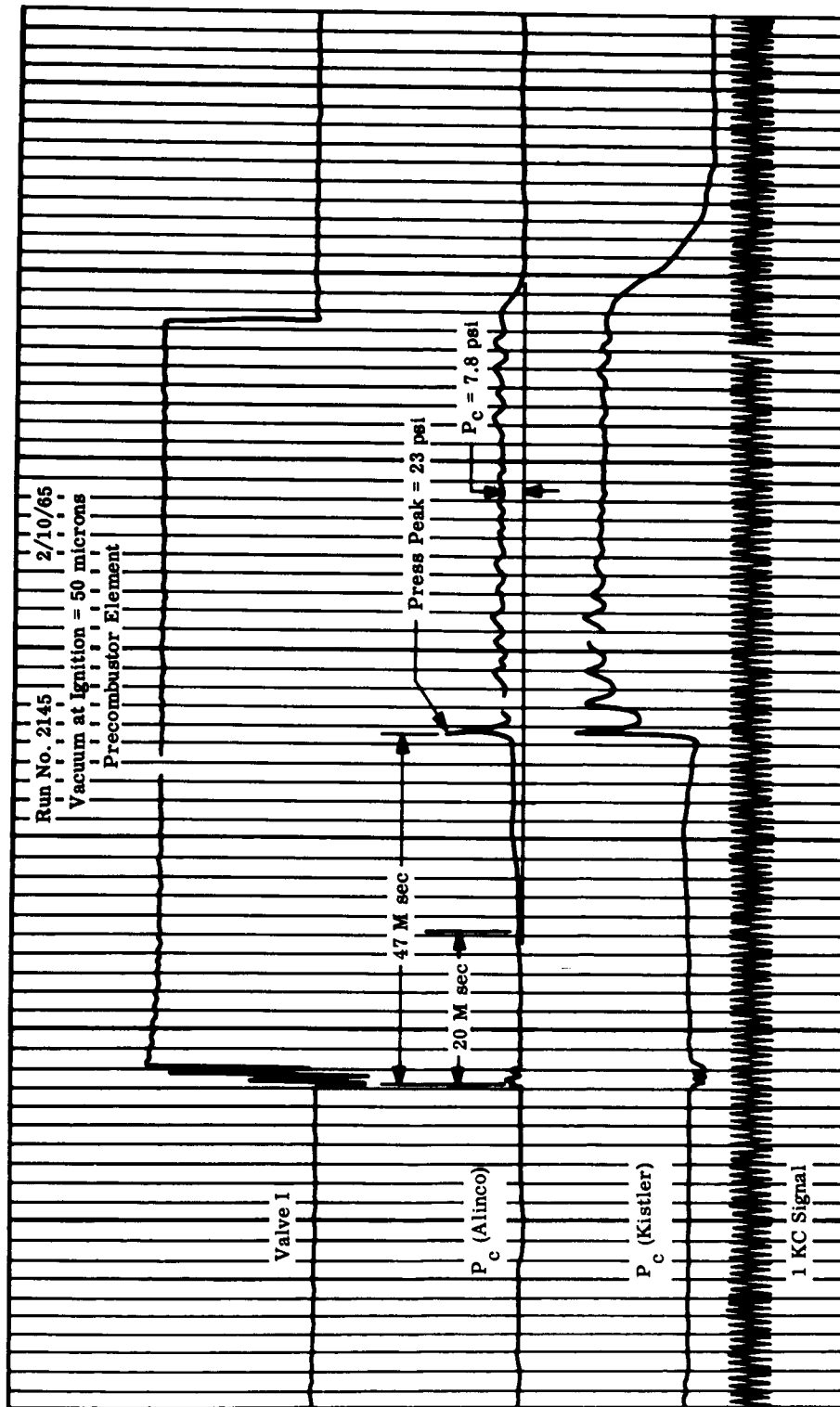


Figure V-97. Oscilloscope Trace - Typical Precombustor Element

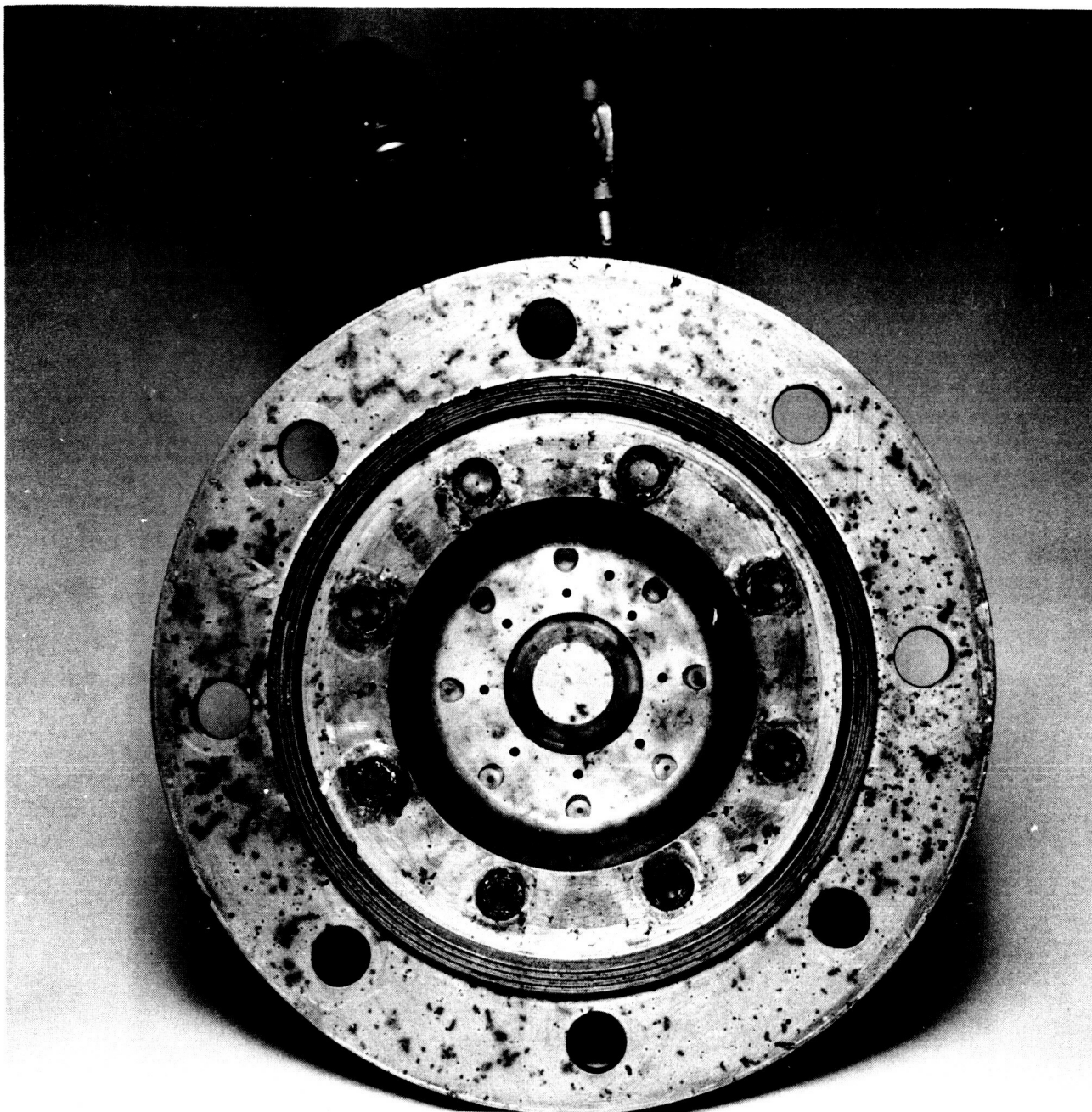
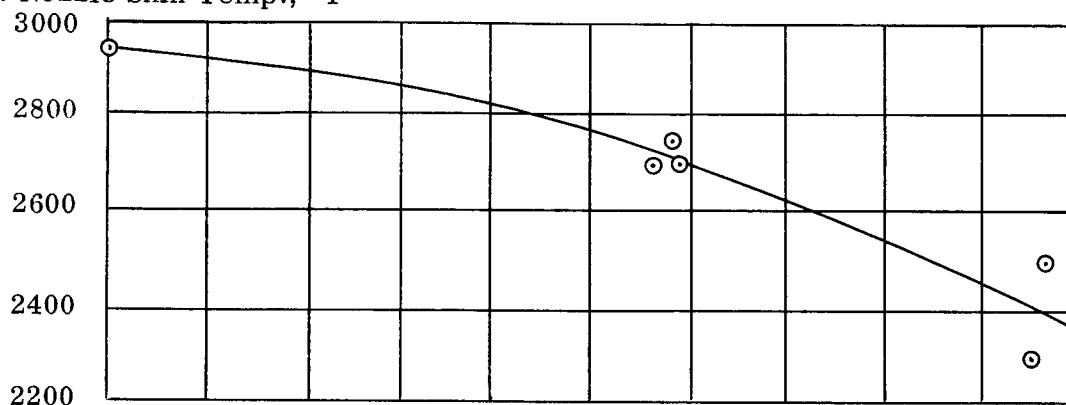


Figure V-93. Film Cooled Injector TF-2A (After Rework)

Model 8374
N₂O₄/MMH

Injector TF-2
O/F - 1.6 P_c - 80 psia

Avg. Nozzle Skin Temp., °F



Characteristic Velocity, ft/sec (corrected)

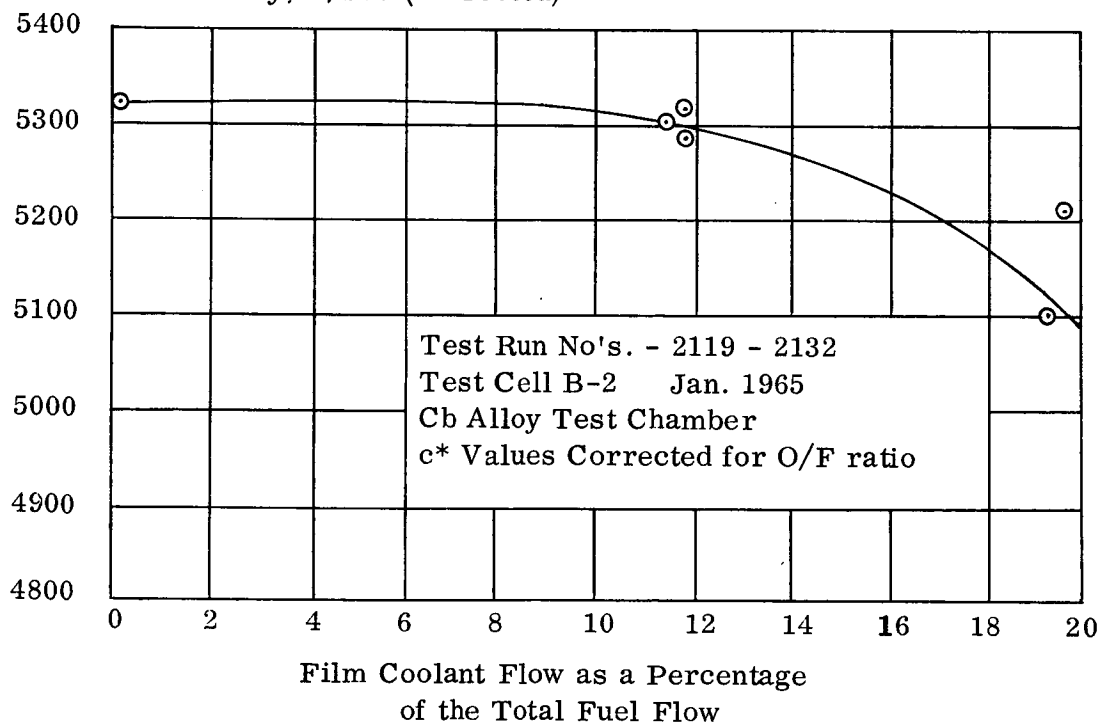


Figure V-99. Characteristic Velocity and Nozzle Skin Temperature as a Function of Film Coolant Flow

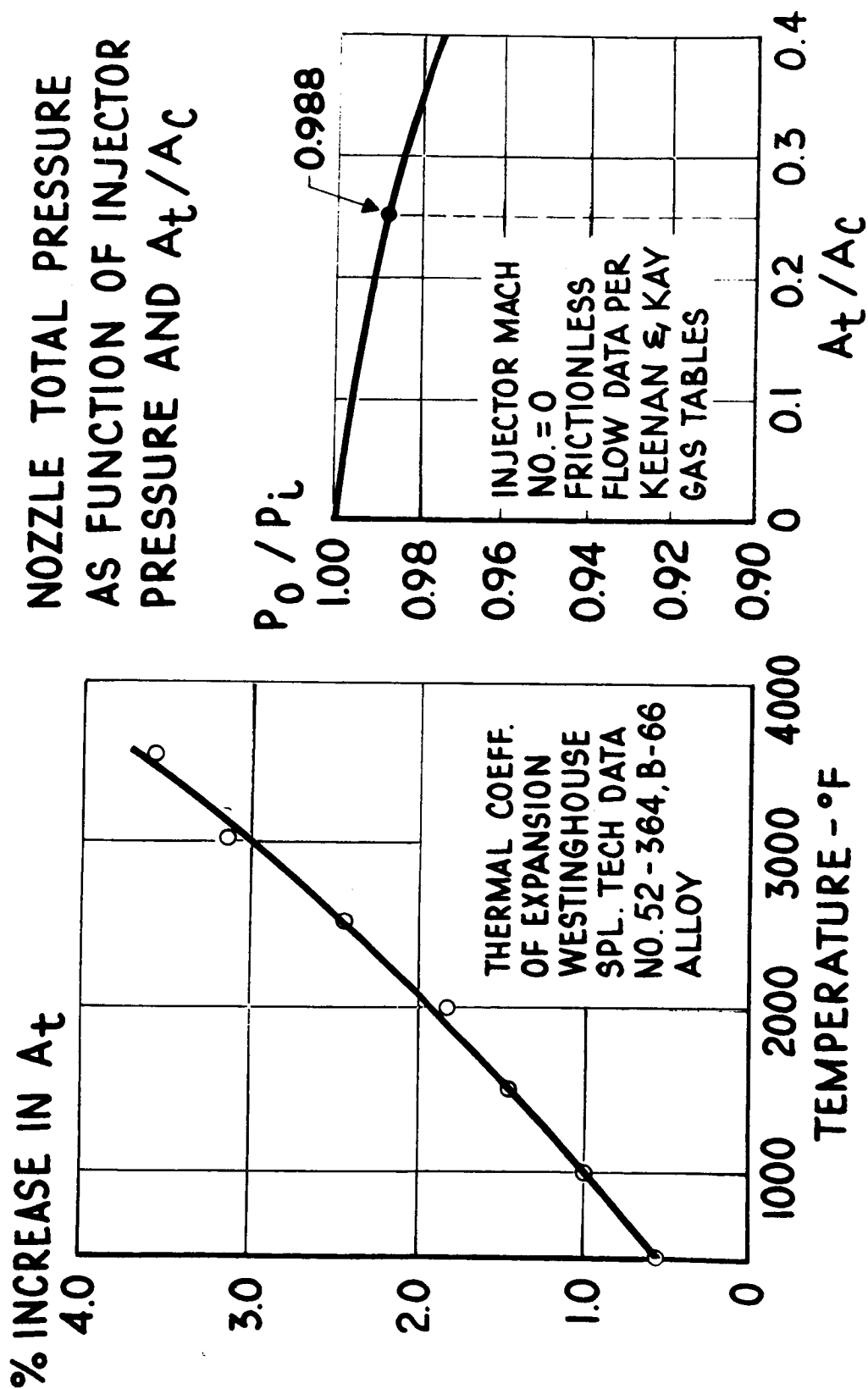


Figure V-100. Correction Factors

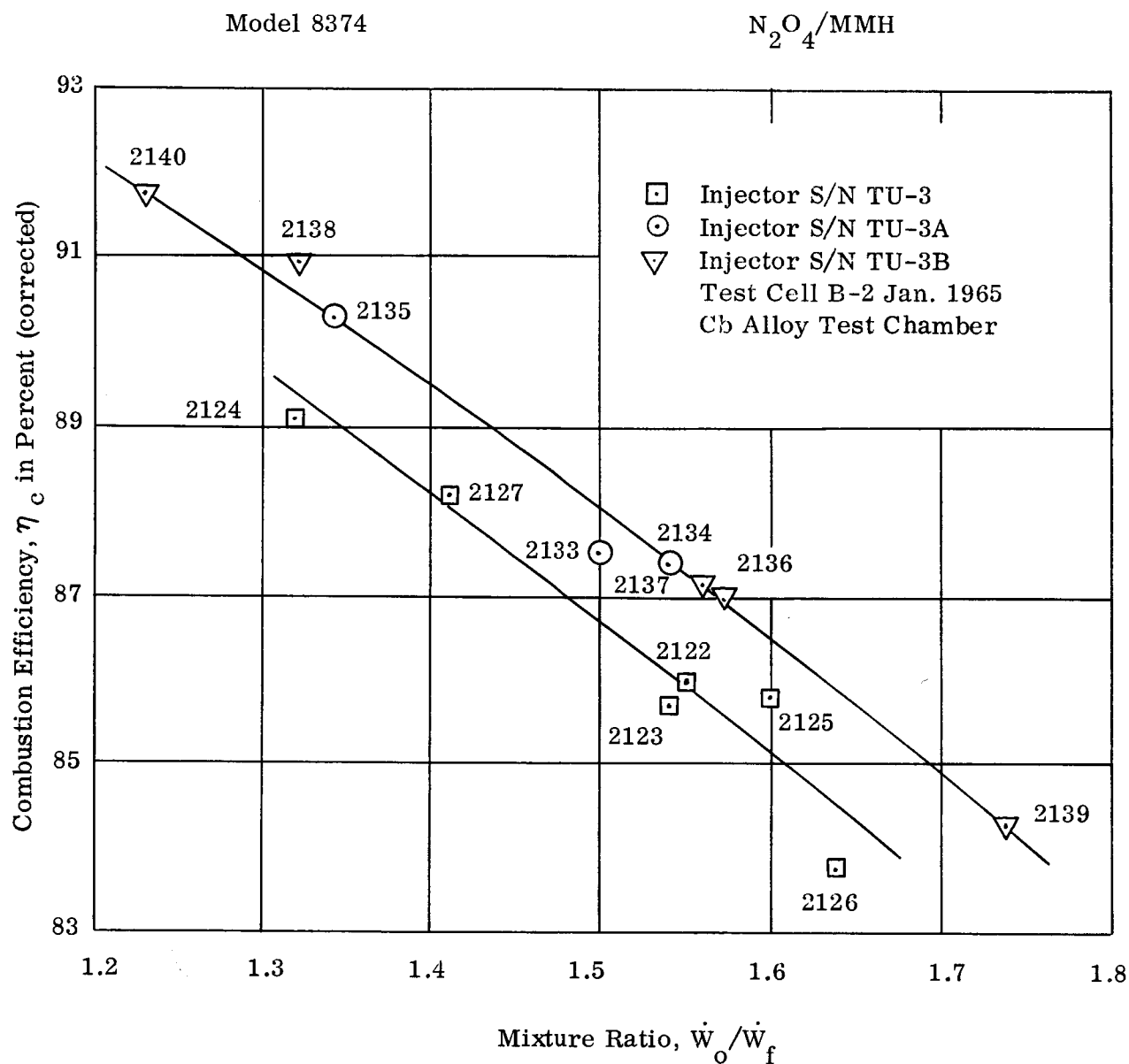


Figure V-101. Combustion Efficiency versus Mixture Ratio

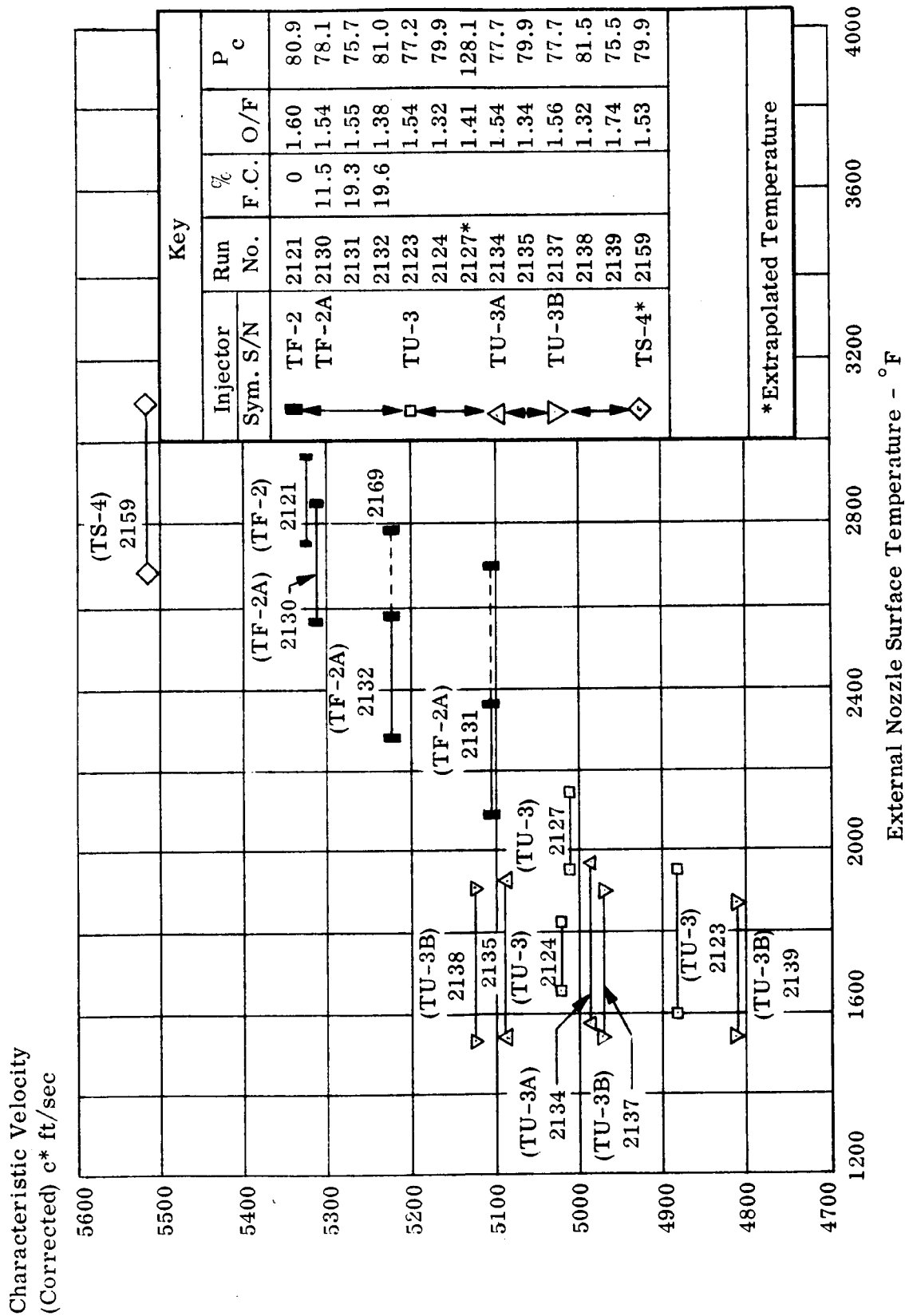


Figure V-102. Model 8374 Nozzle Temperature Data for Various Stabilized Operating Conditions

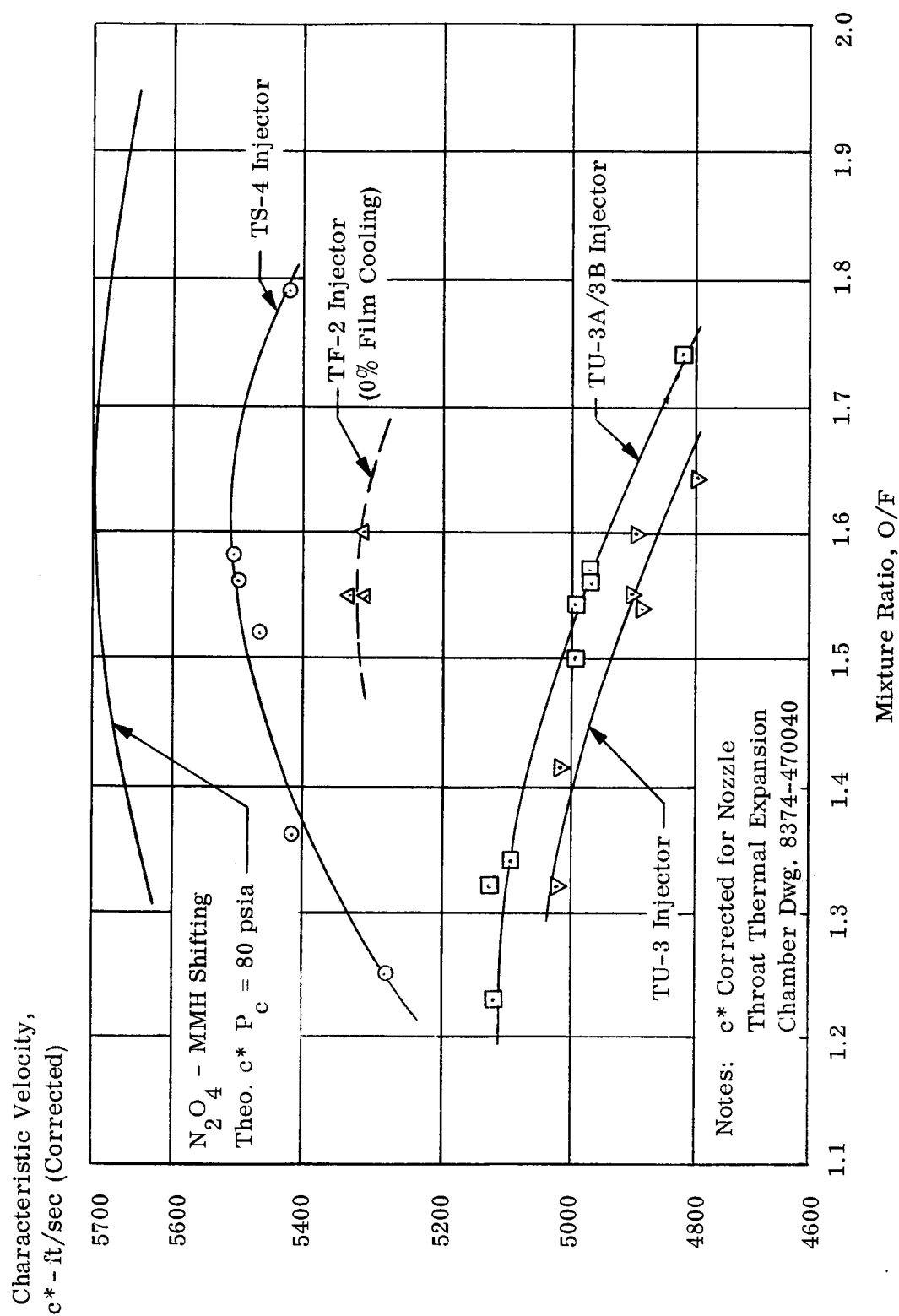
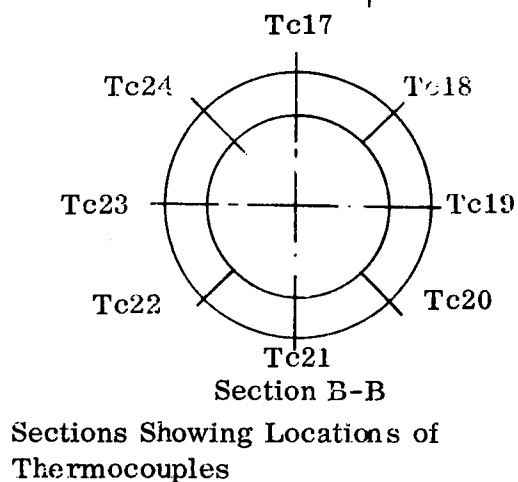
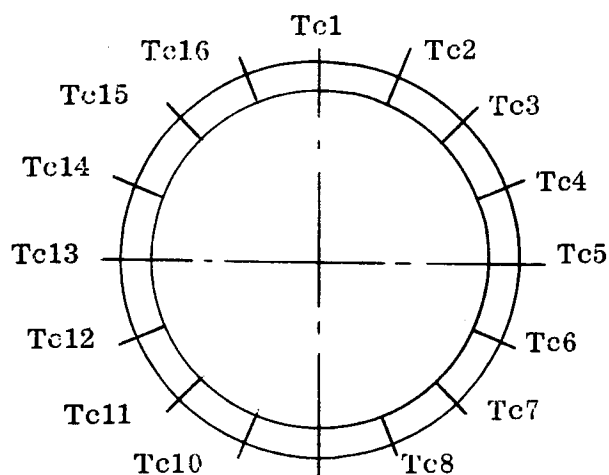
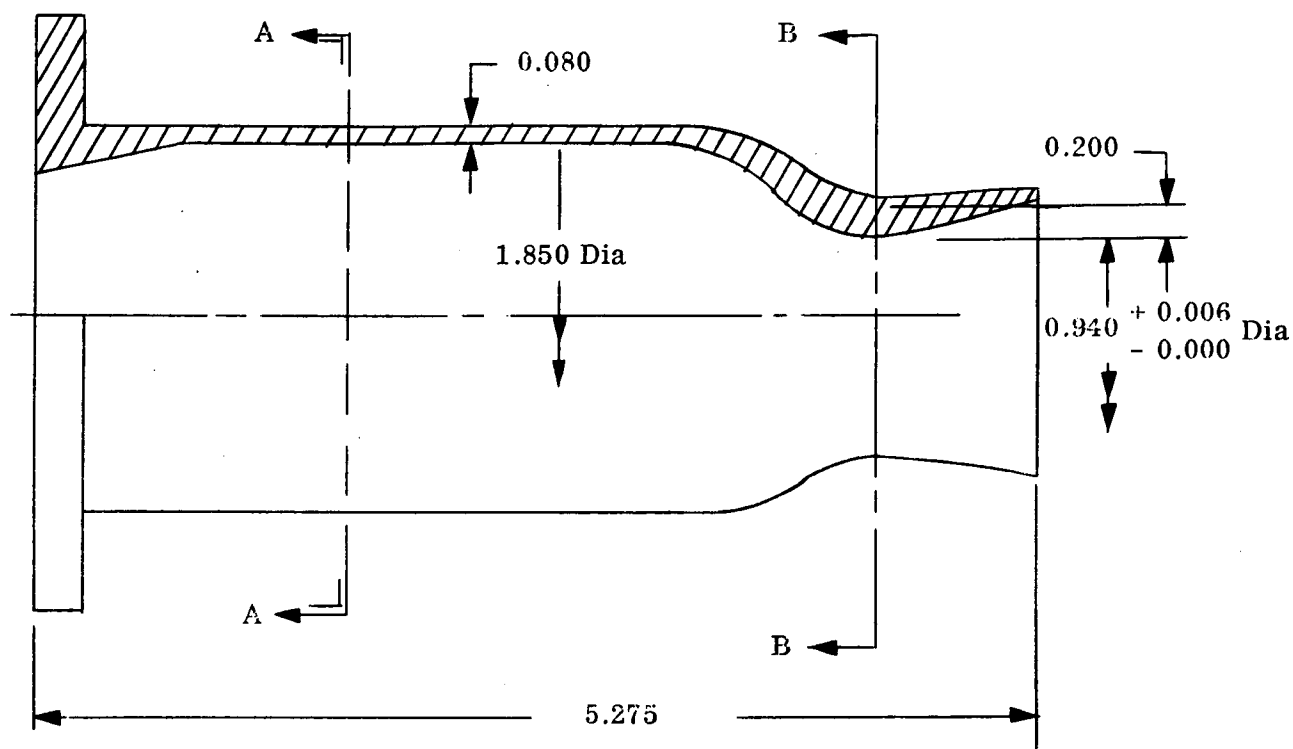


Figure V-103. Injector Performance in a Hot-Wall Chamber



Mat'l - Ch - 1 Zr
Coating - Silicide
Nominal $P_c = 80$ psia
Thermocouple - Pt/Pt-Rh

Ref Dwg 8374-470040

Figure V-104. COLUMBIUM CHAMBER - Thermocouple Location

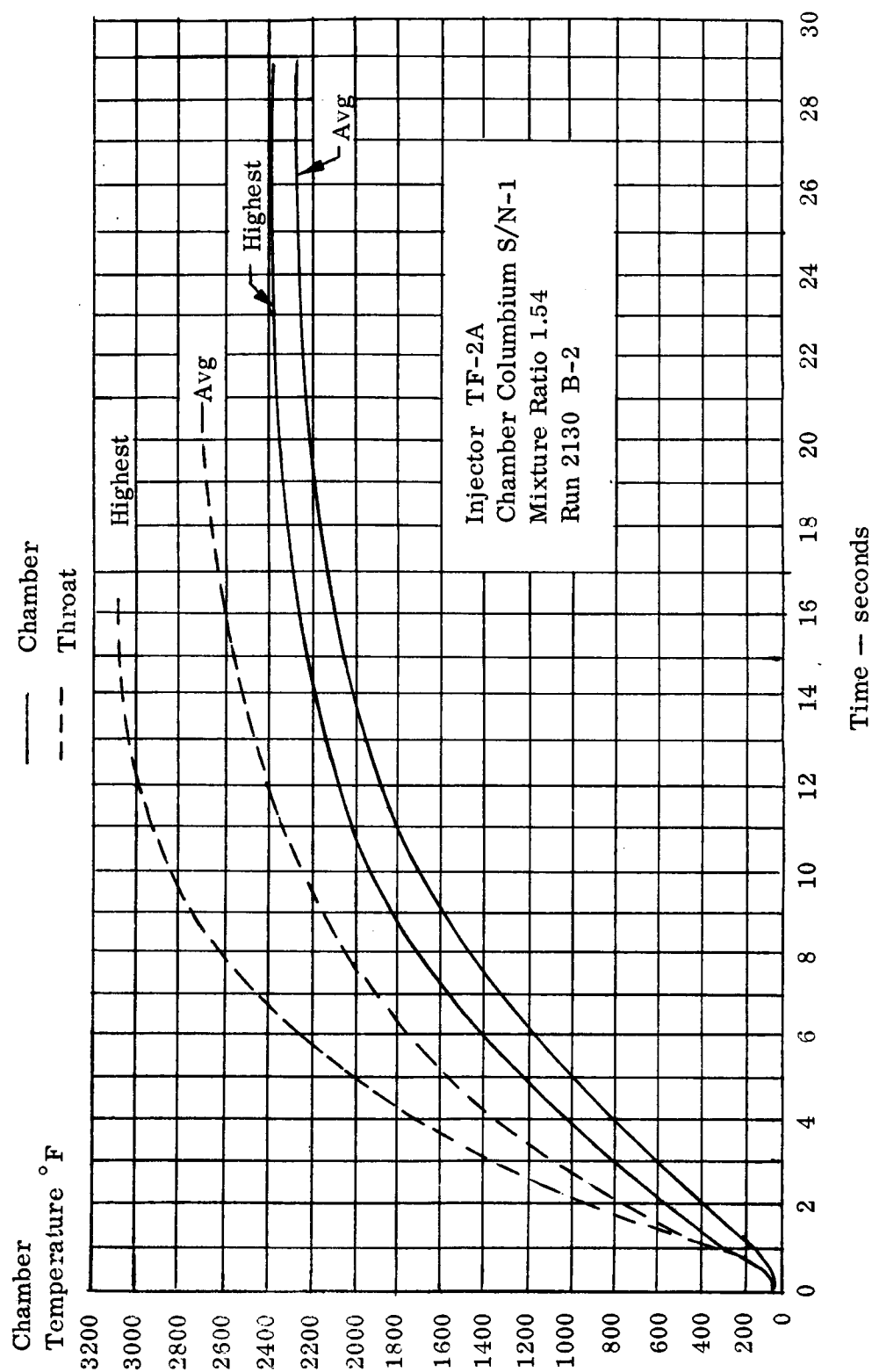


Figure V-105. Chamber Temperature versus Time

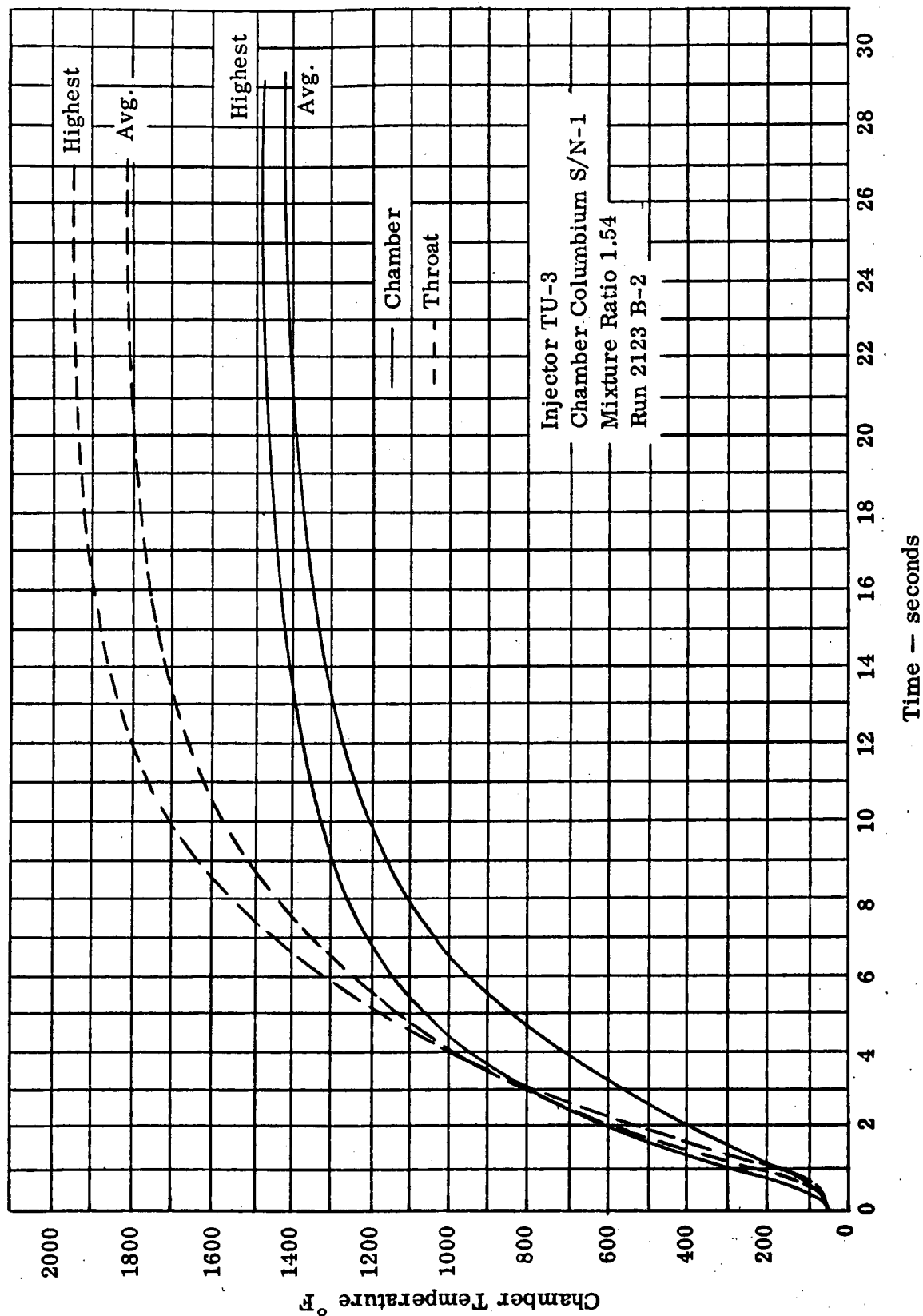


Figure V-106. Chamber Temperature versus Time

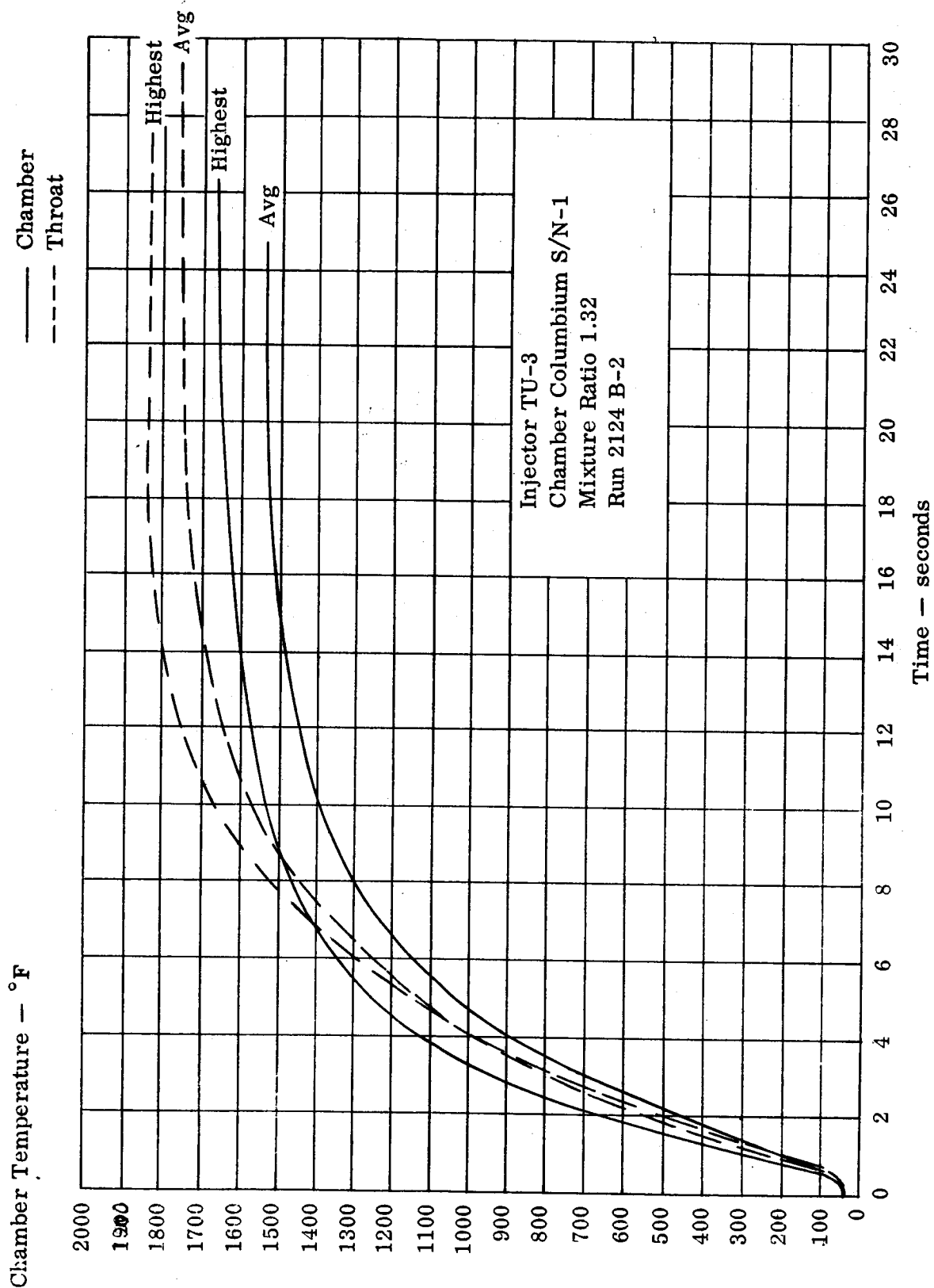


Figure V-107. Chamber Temperature versus Time

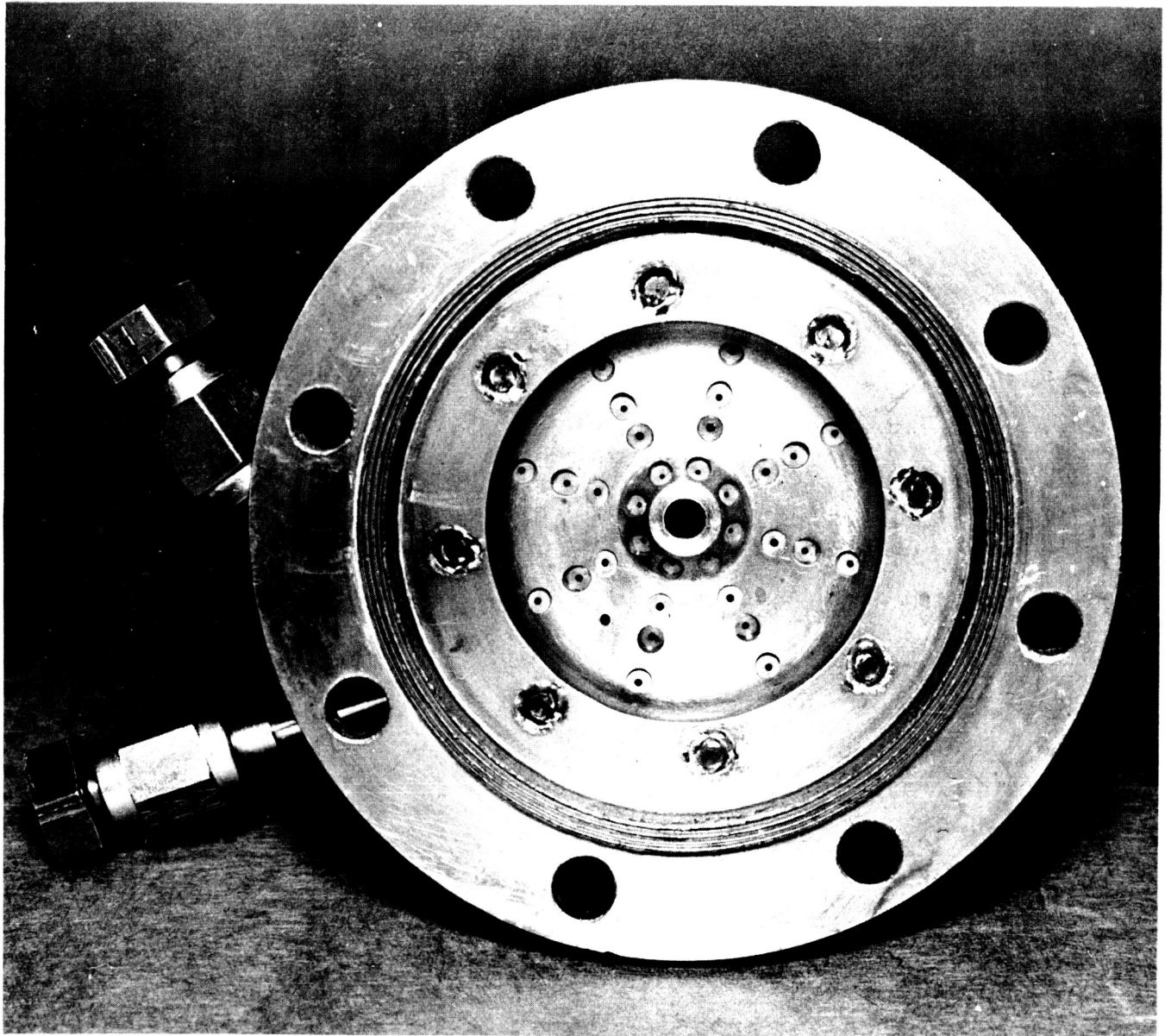


Figure V-108. Injector TF-5

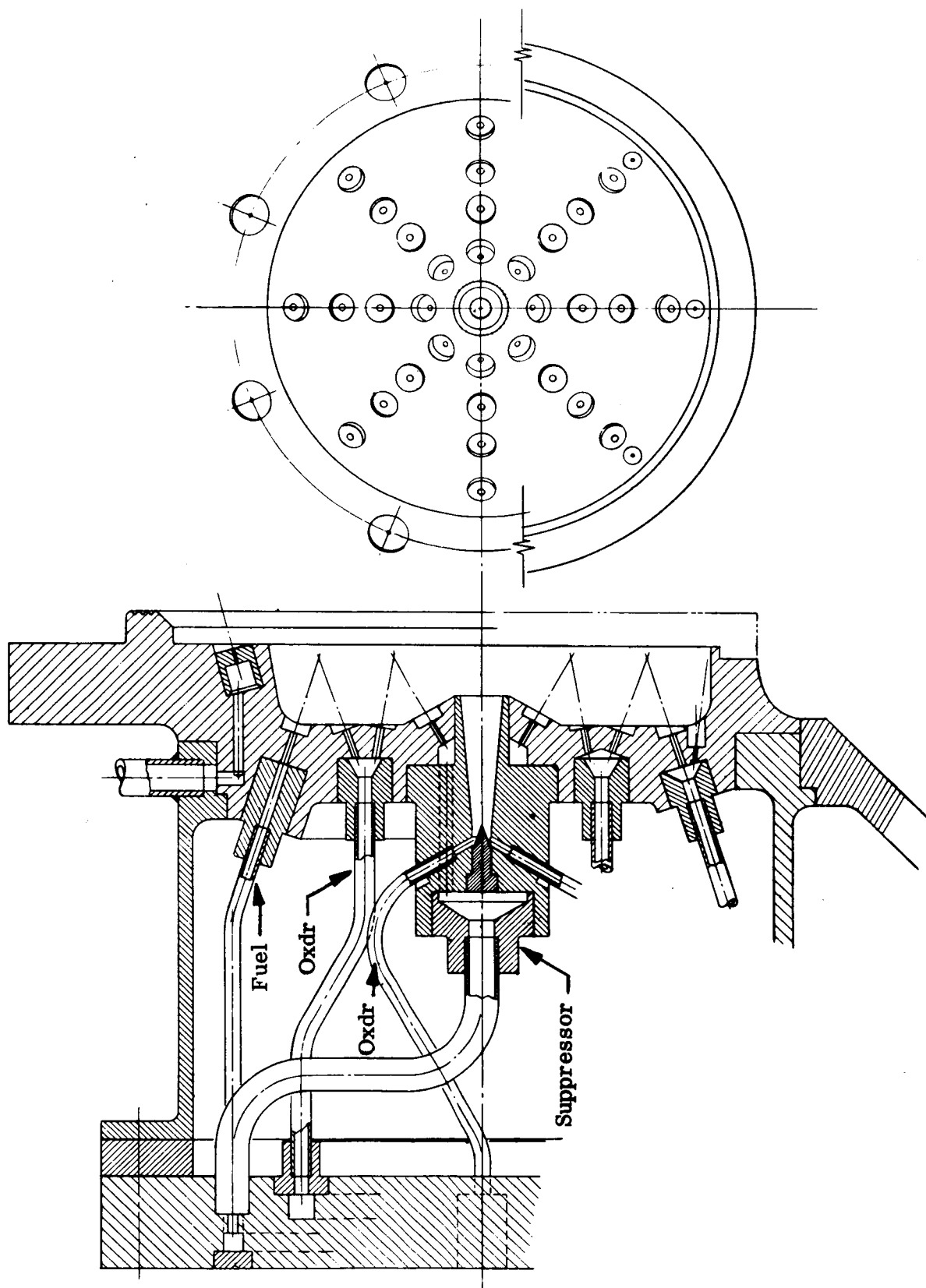


Figure V-109. Injector TF-5

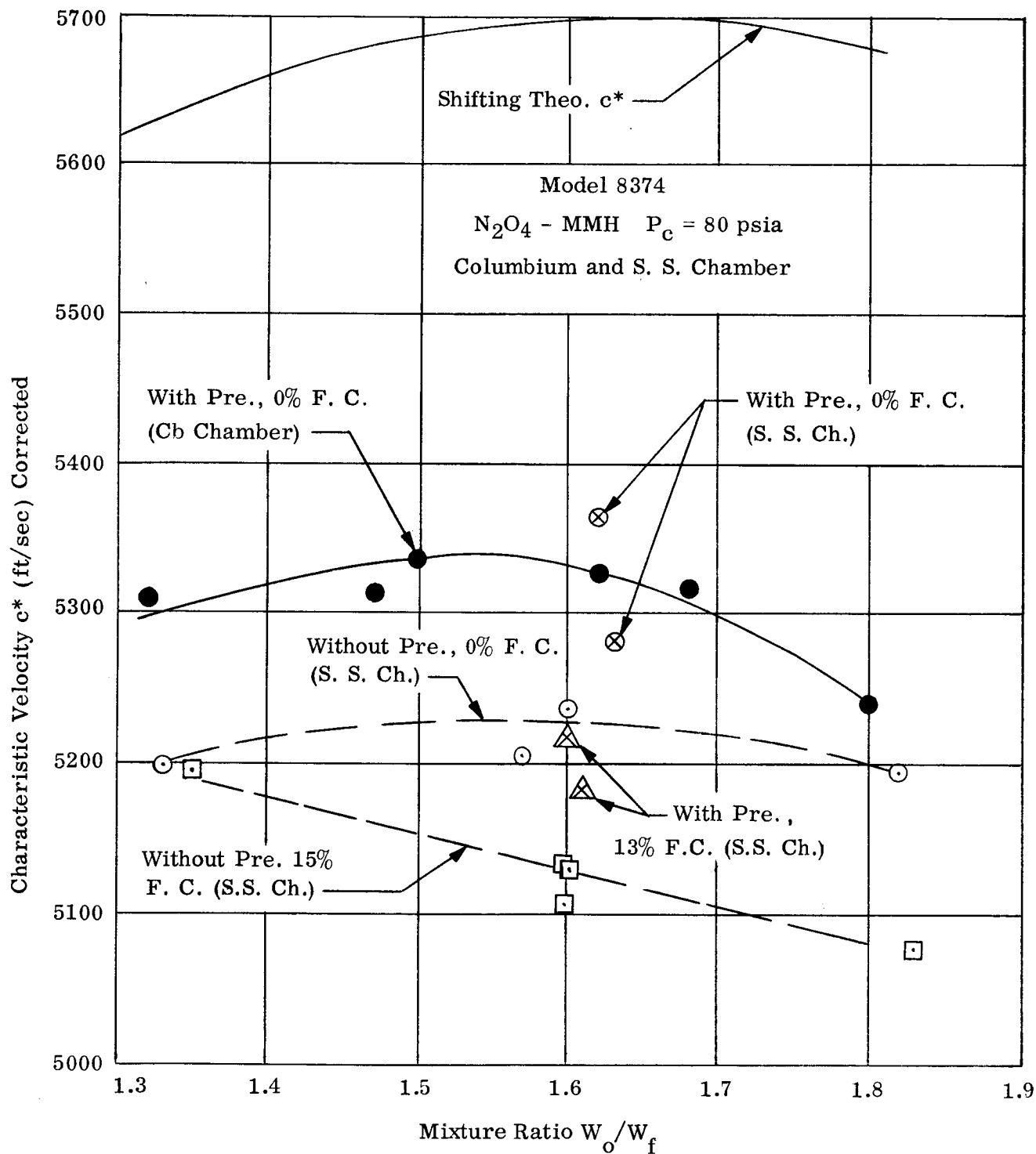


Figure V-110. Characteristic Velocity versus Mixture Ratio Injector TF-5



R-187

Figure V-111. Run 2195 B-2 Sea Level

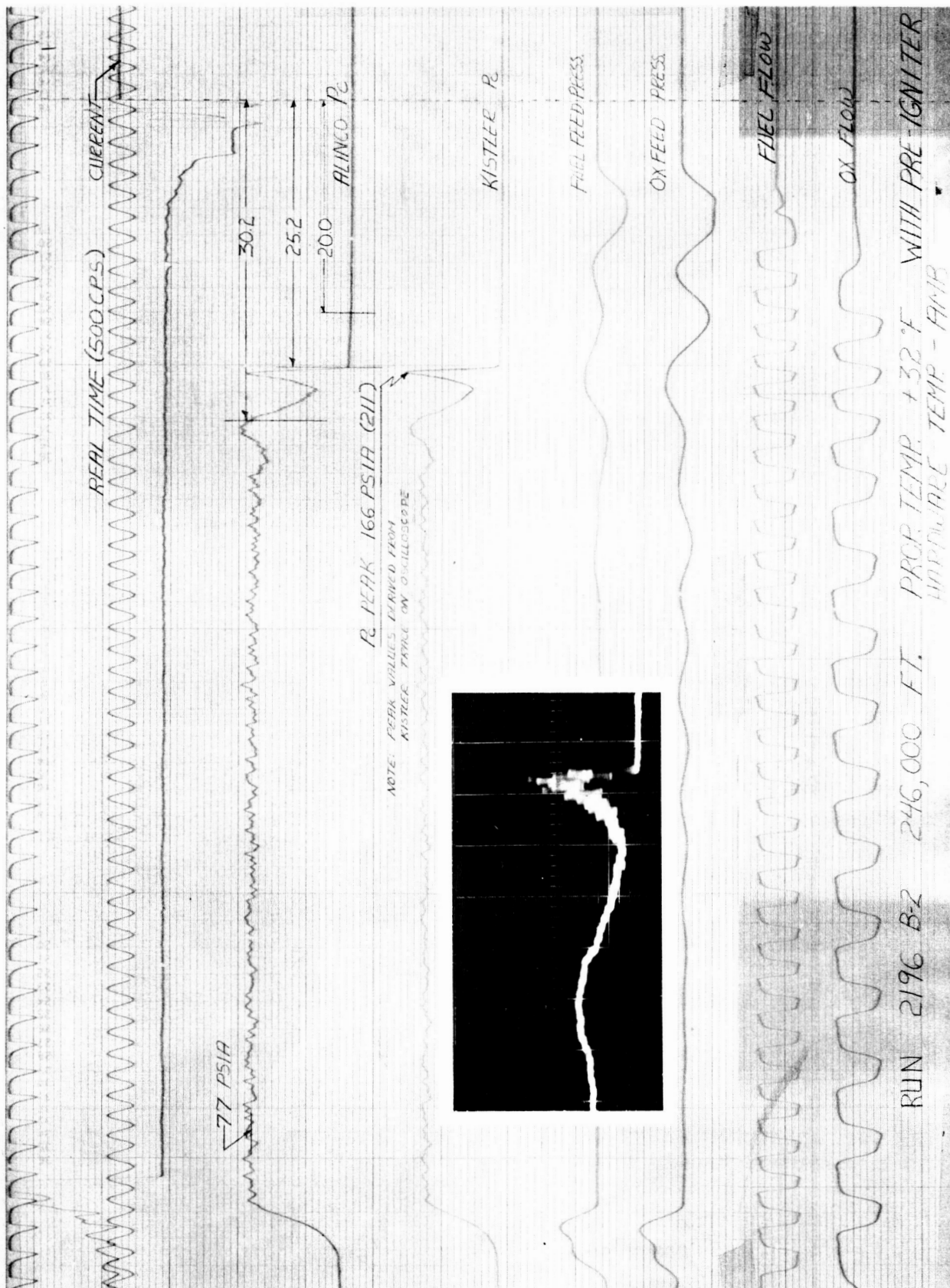


Figure V-112. Run 2196 B-2 246,000 ft

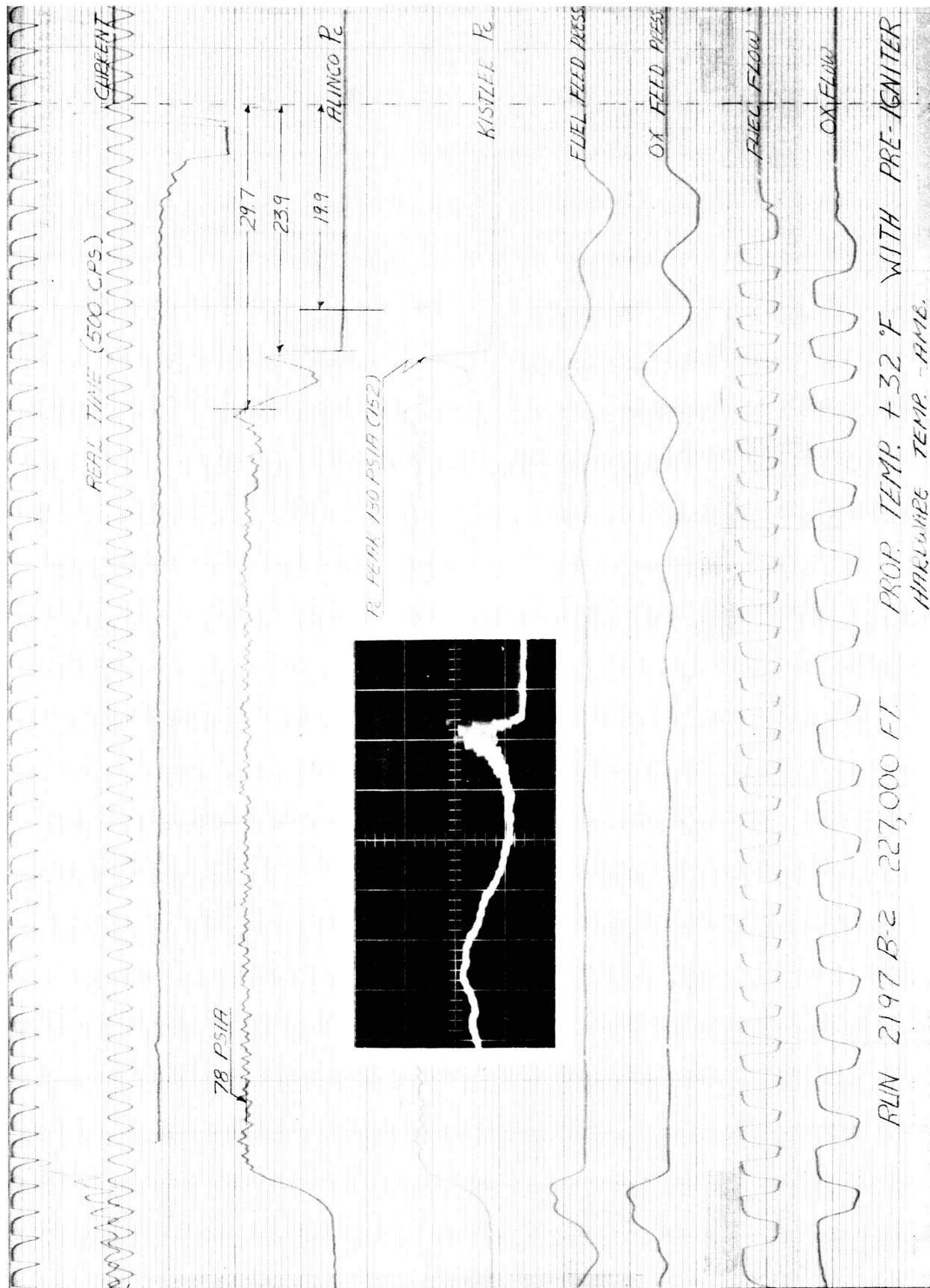


Figure V-113. Run 2197 B-2 227,000 ft

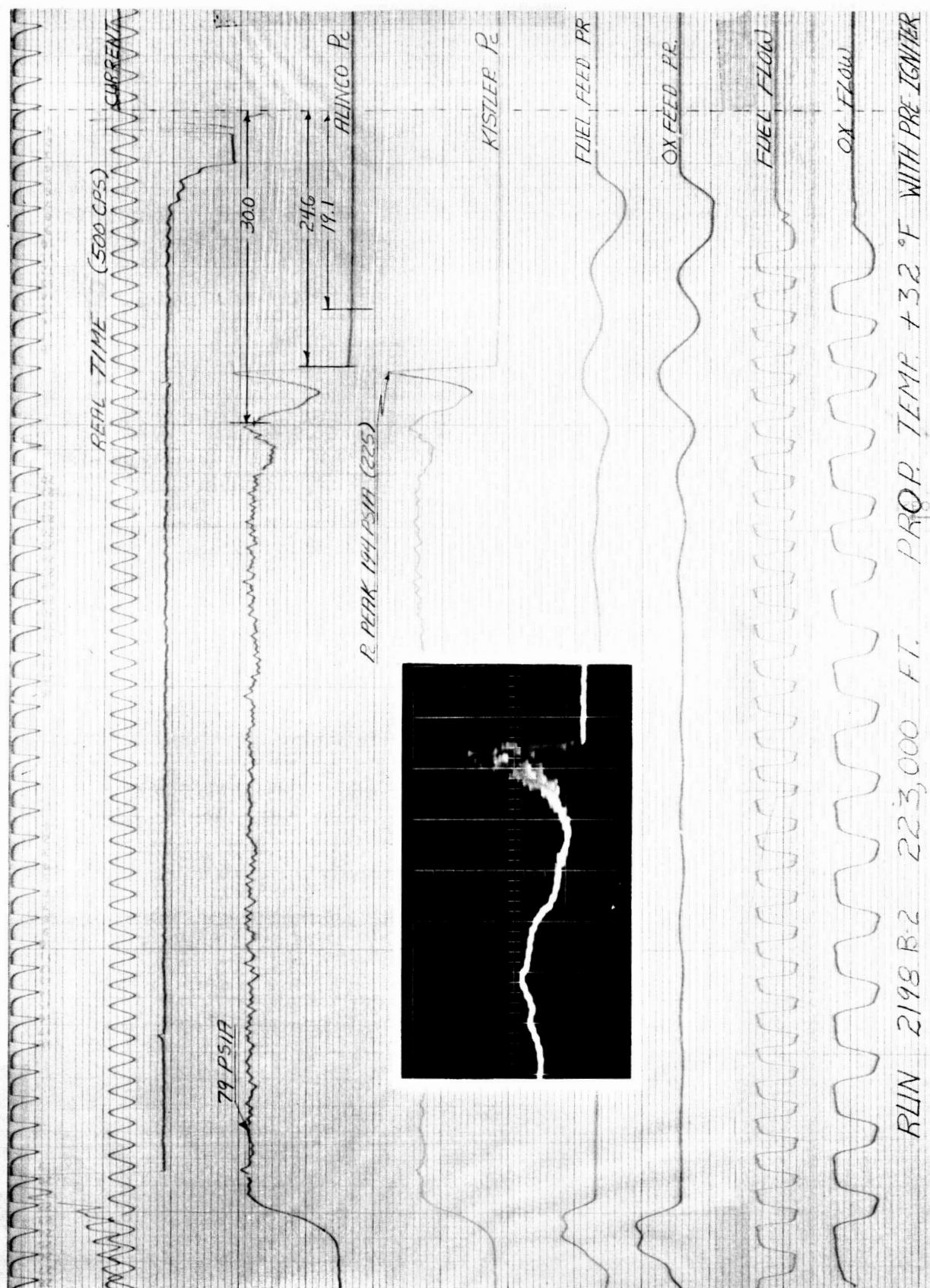


Figure V-114. Run 2198 V-2 223,000 ft

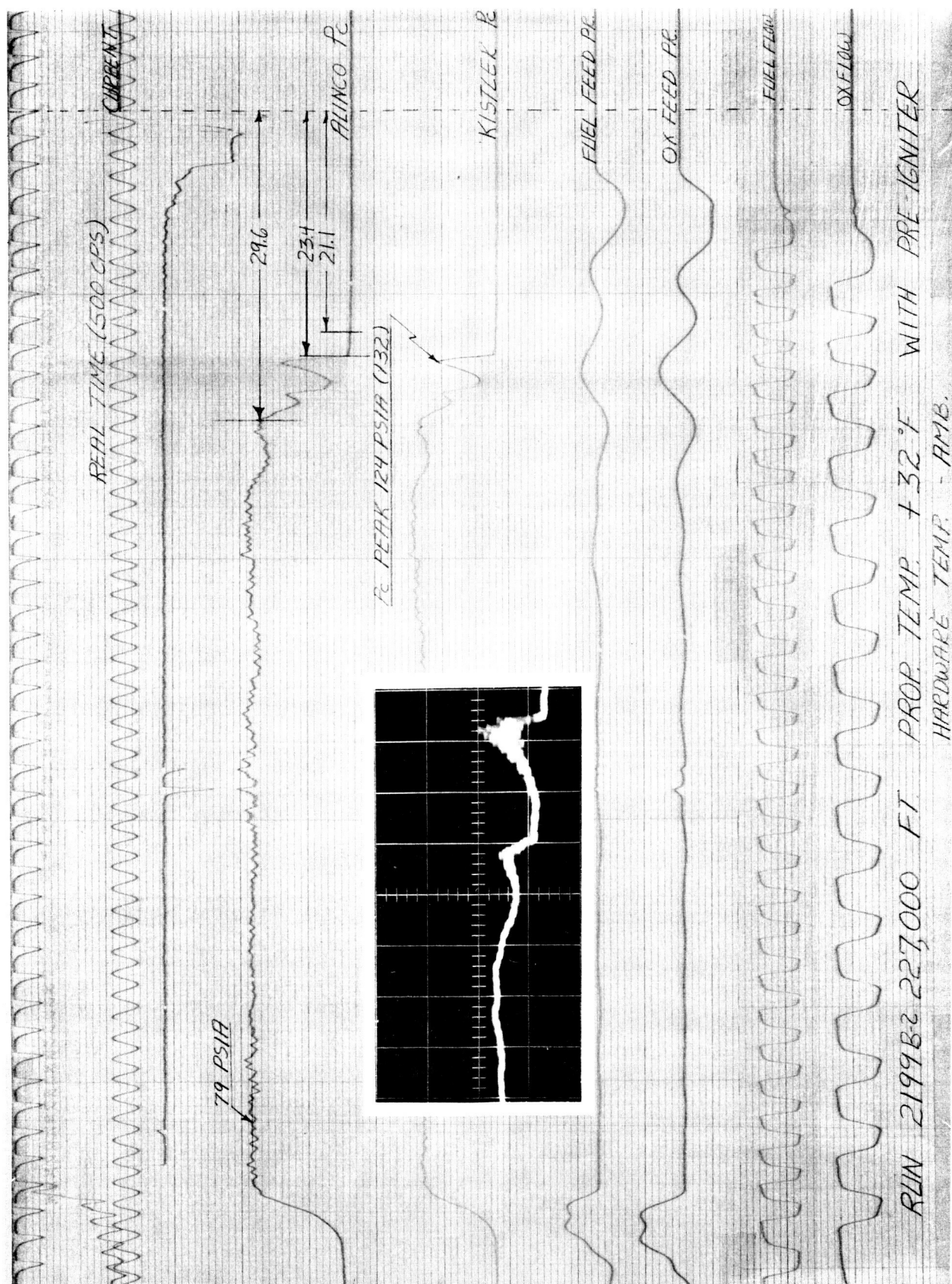


Figure V-115. Run 2199 B-2 227,000 ft

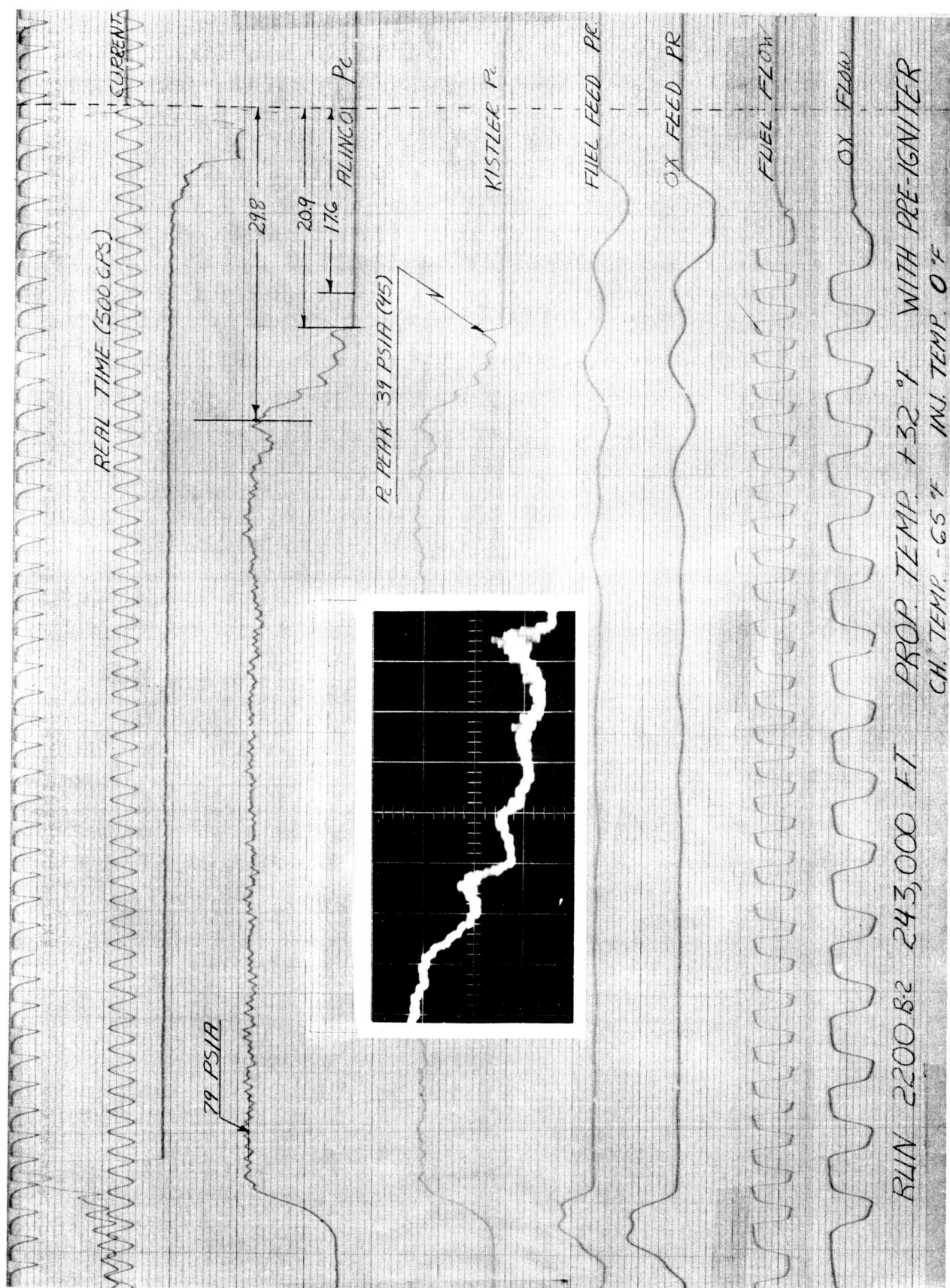


Figure V-116. Run 2200 B-2 243,000 ft

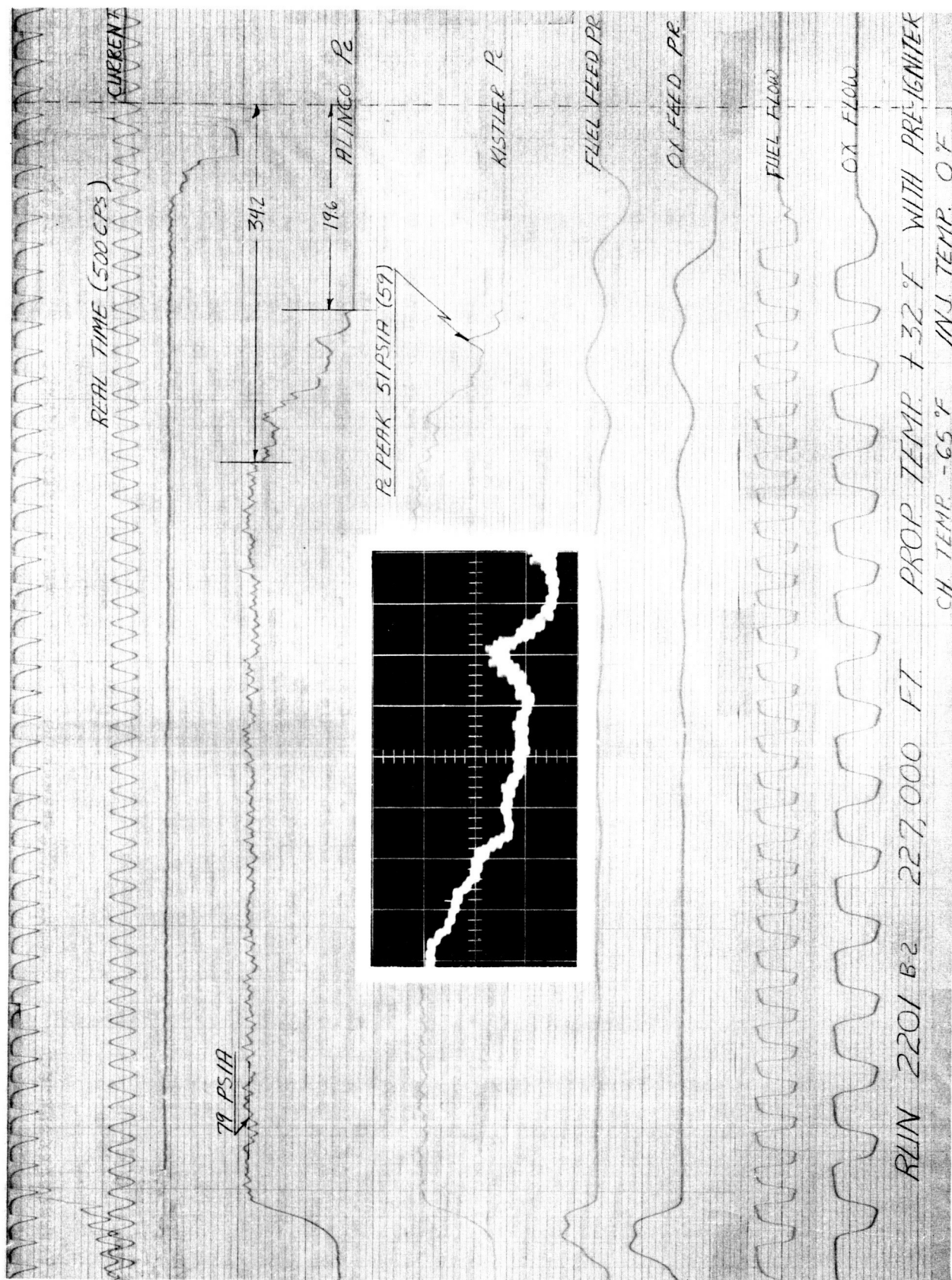


Figure V-117. Run 2201 B-2 227,000 ft

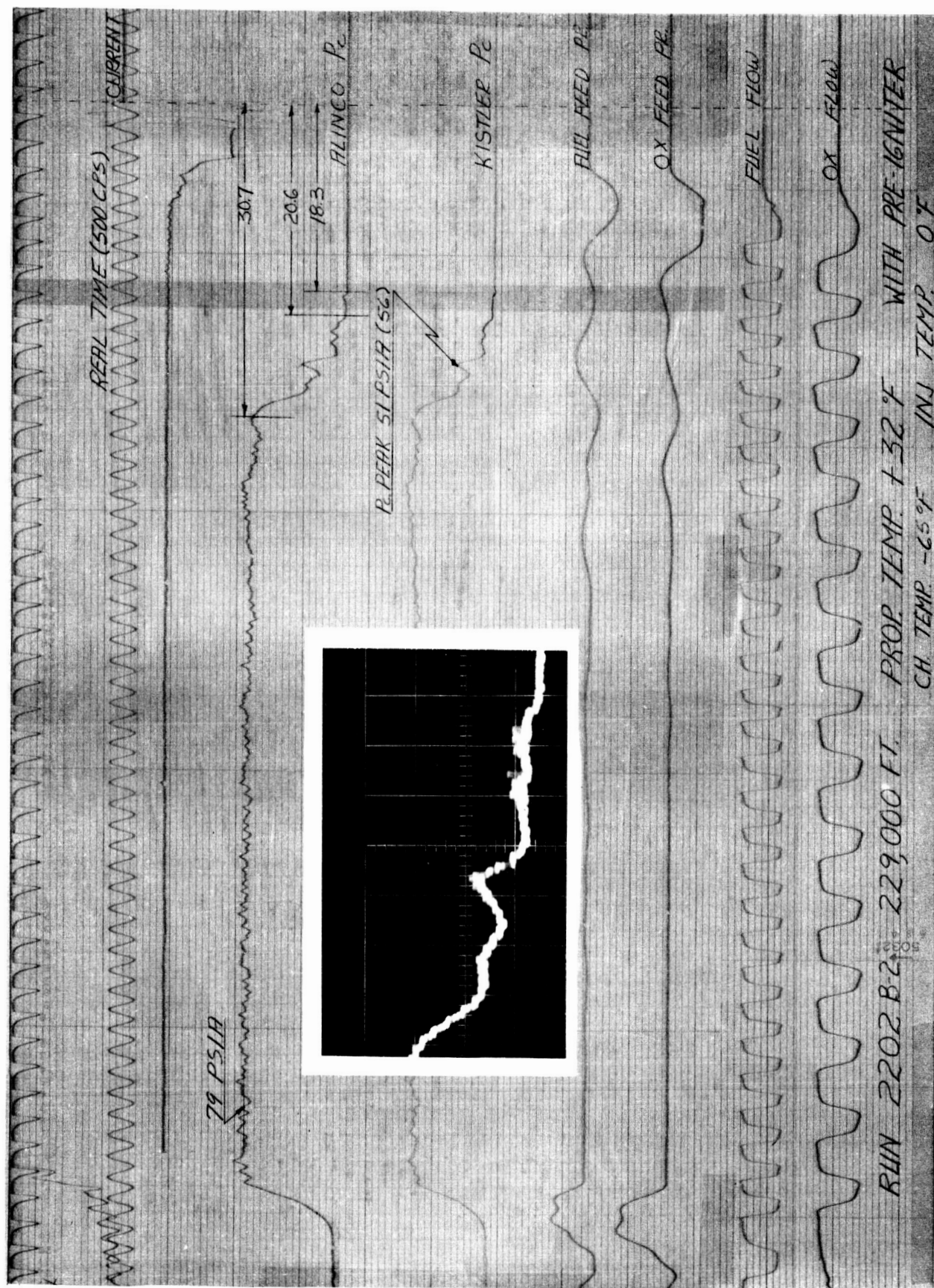


Figure V-118. Run 2202 B-2 229,000 ft

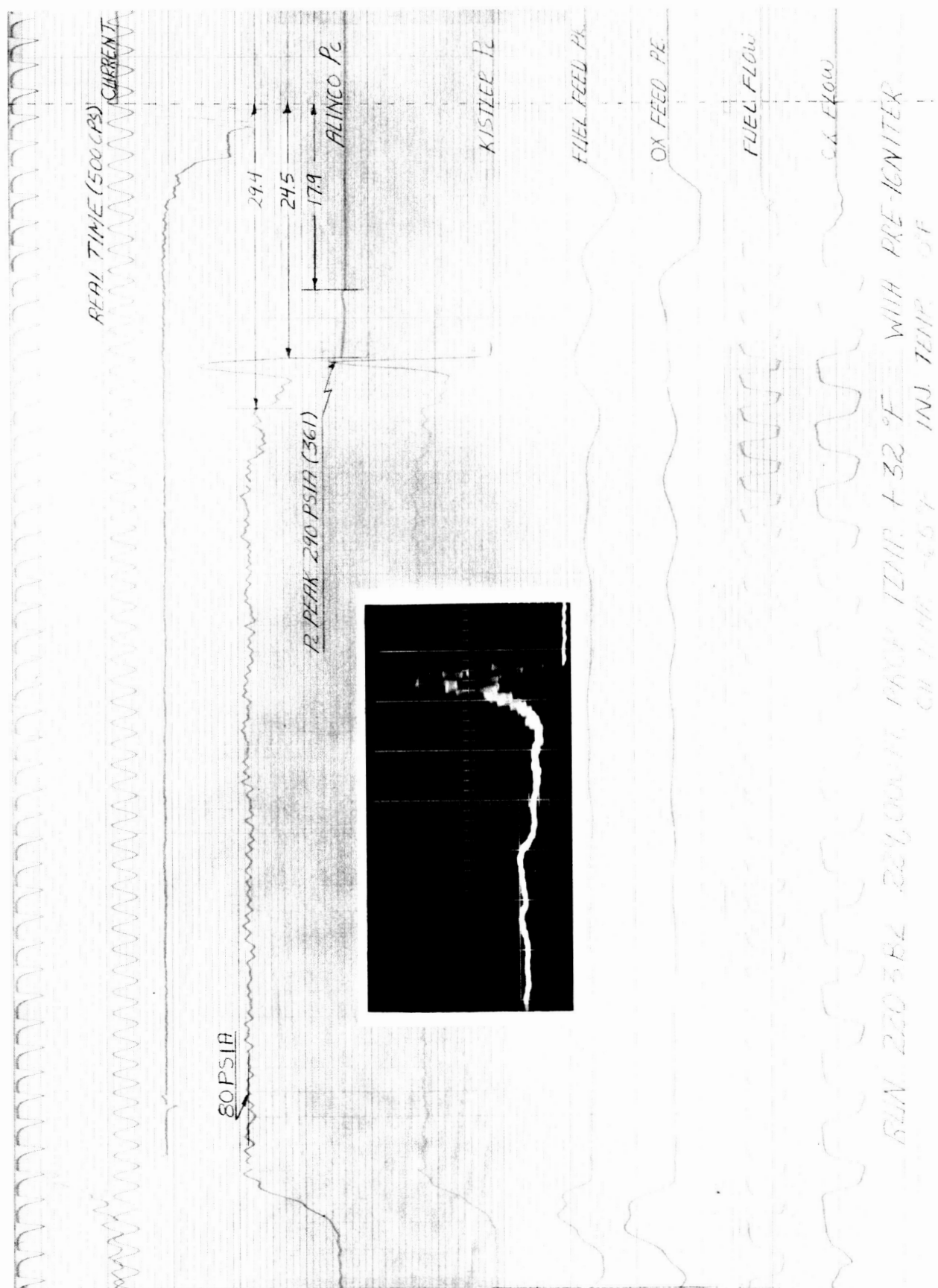


Figure V-119. Run 2203 B-2 229,000 ft

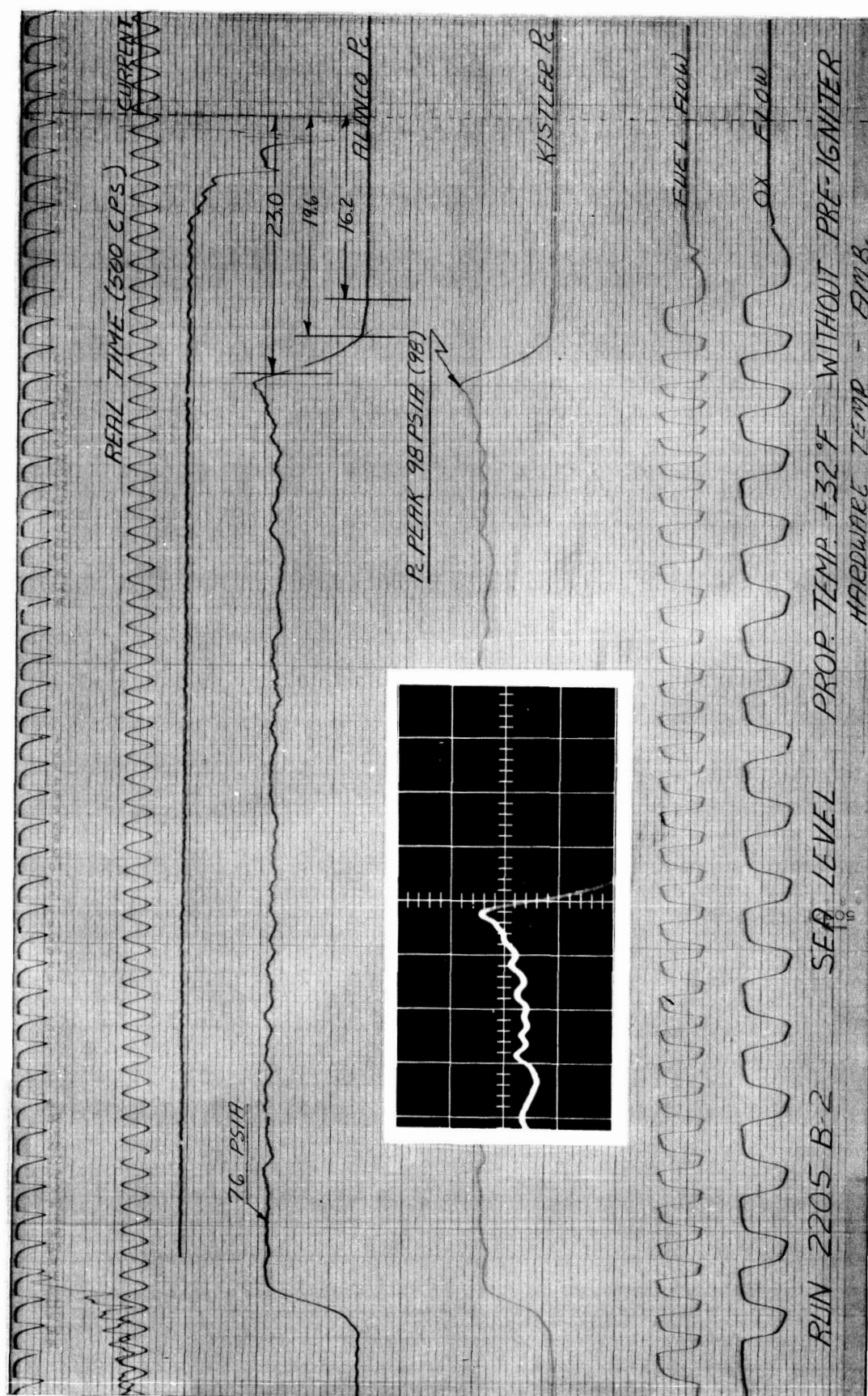


Figure V-120. Run 2205-B-2 - Sea Level

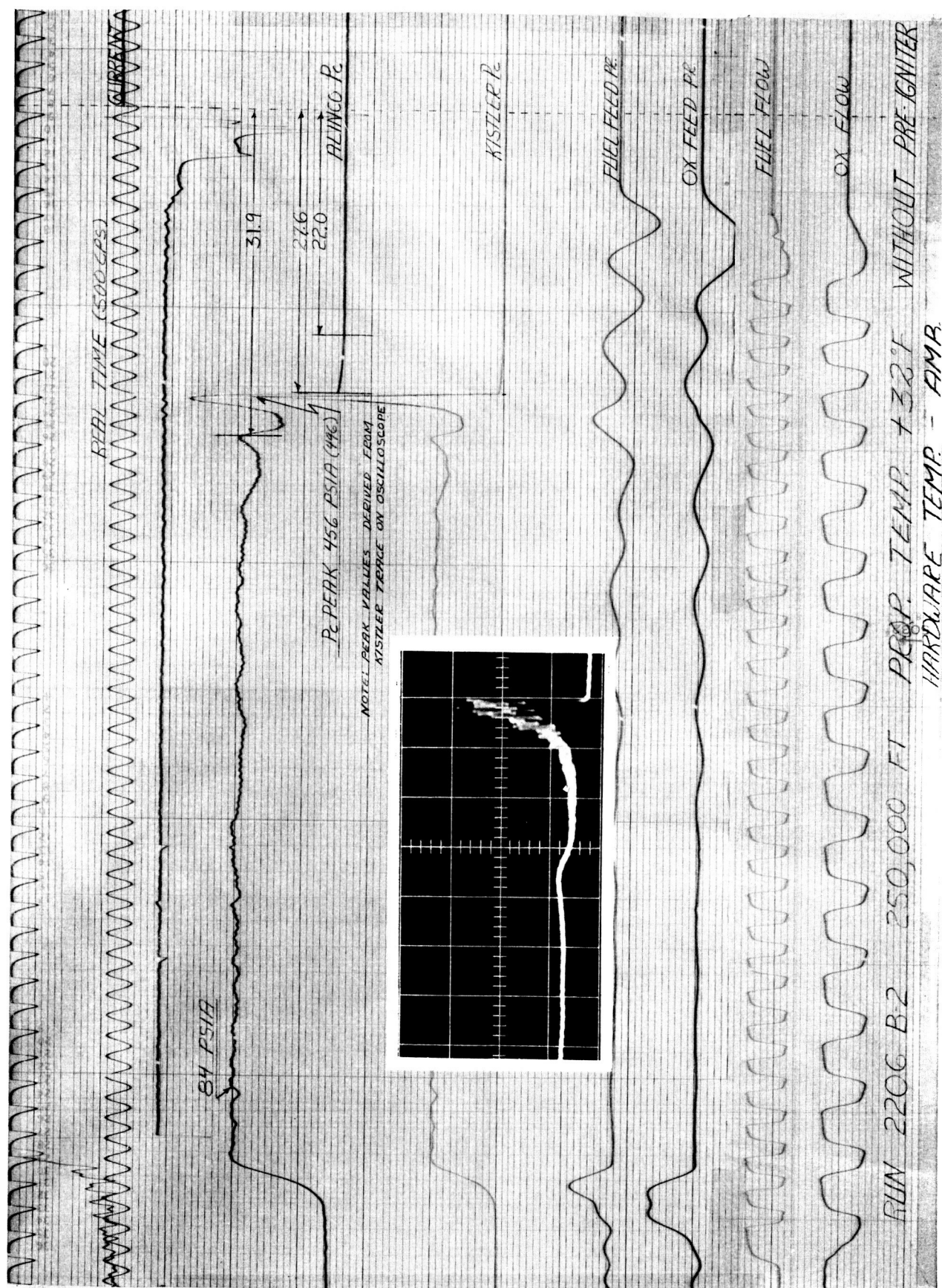


Figure V-121. Run 2206 B-2 at 250,000 Feet

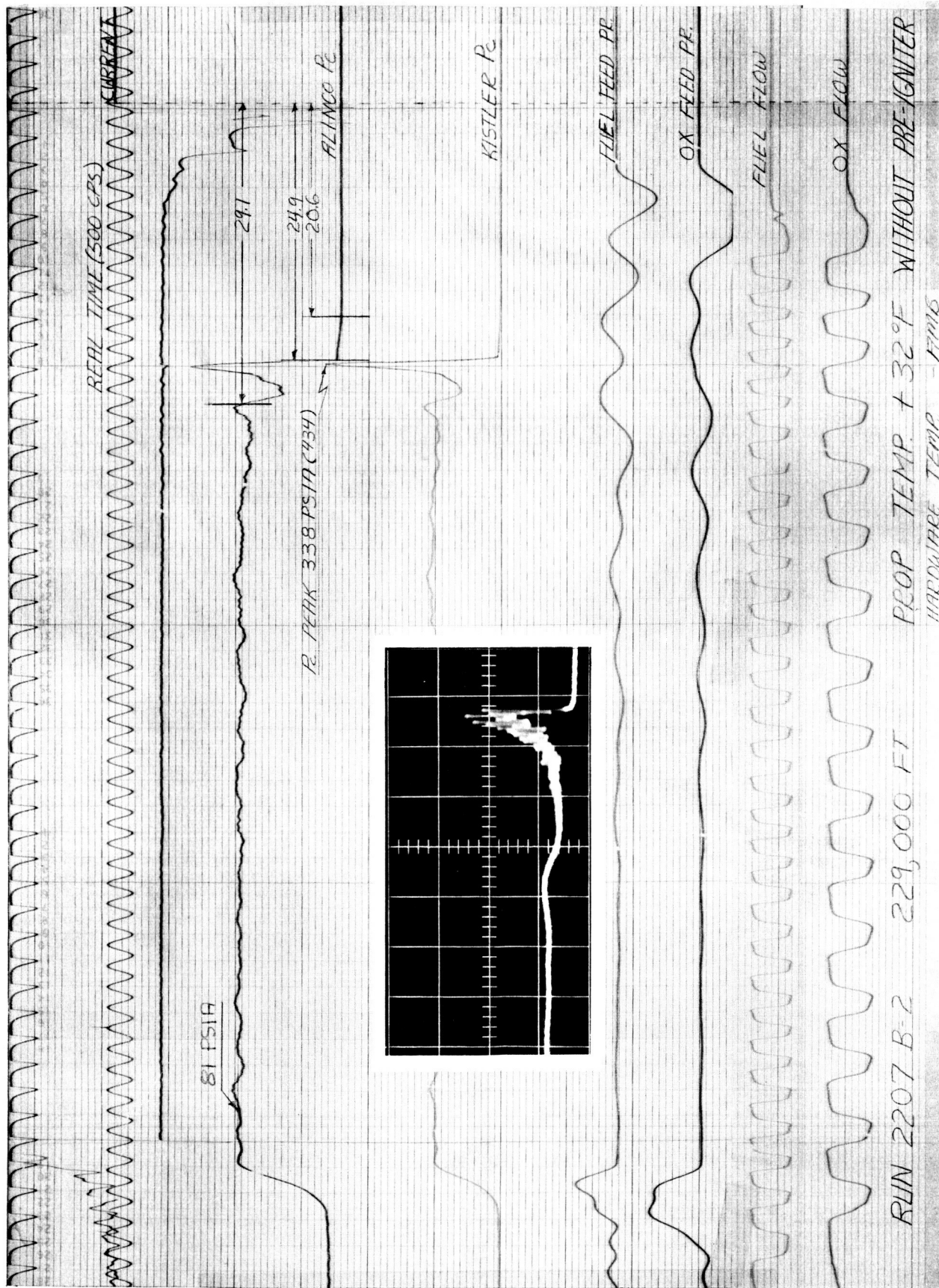


Figure V-122. Run 2207 B-2 229,000 ft

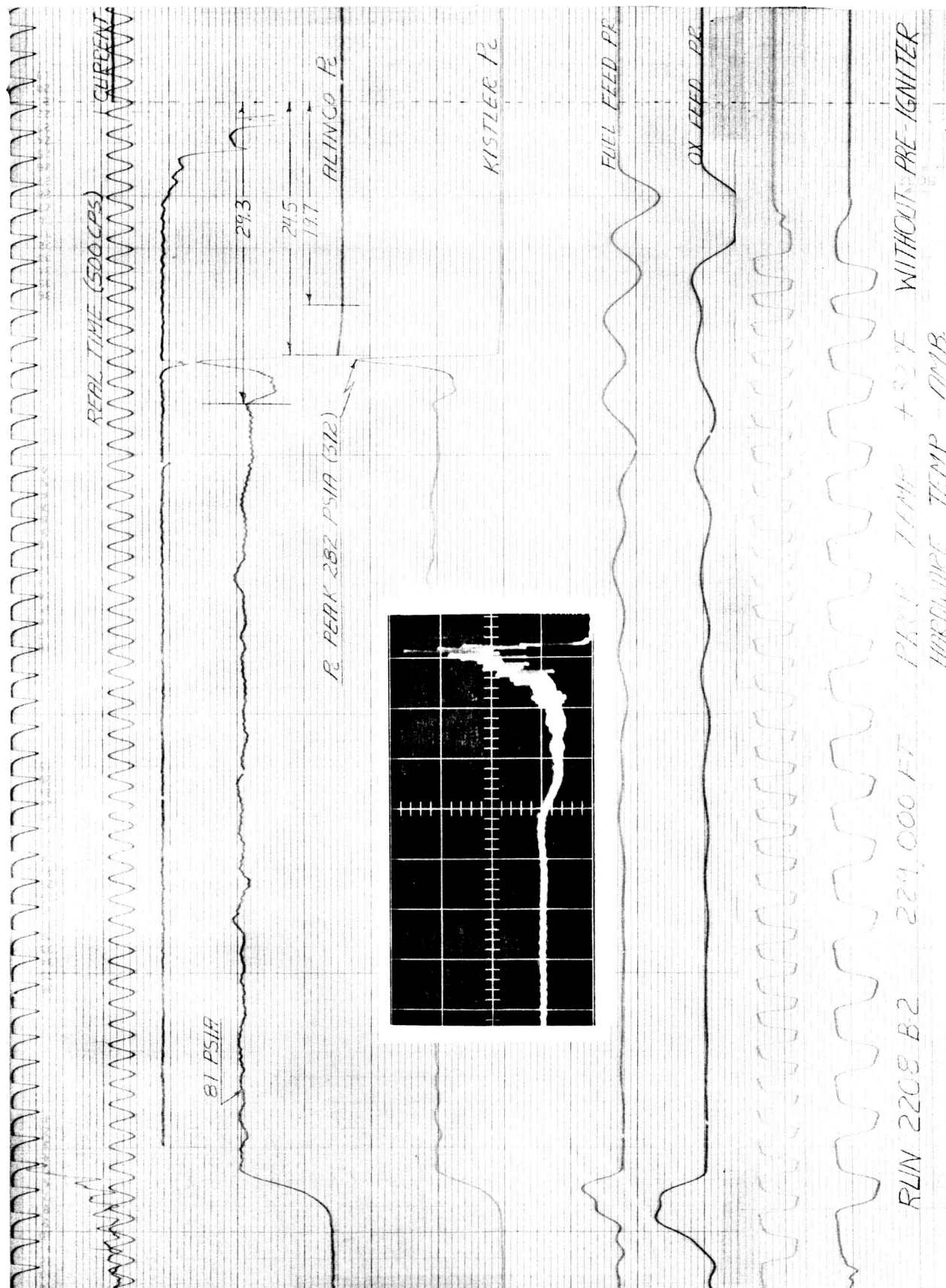
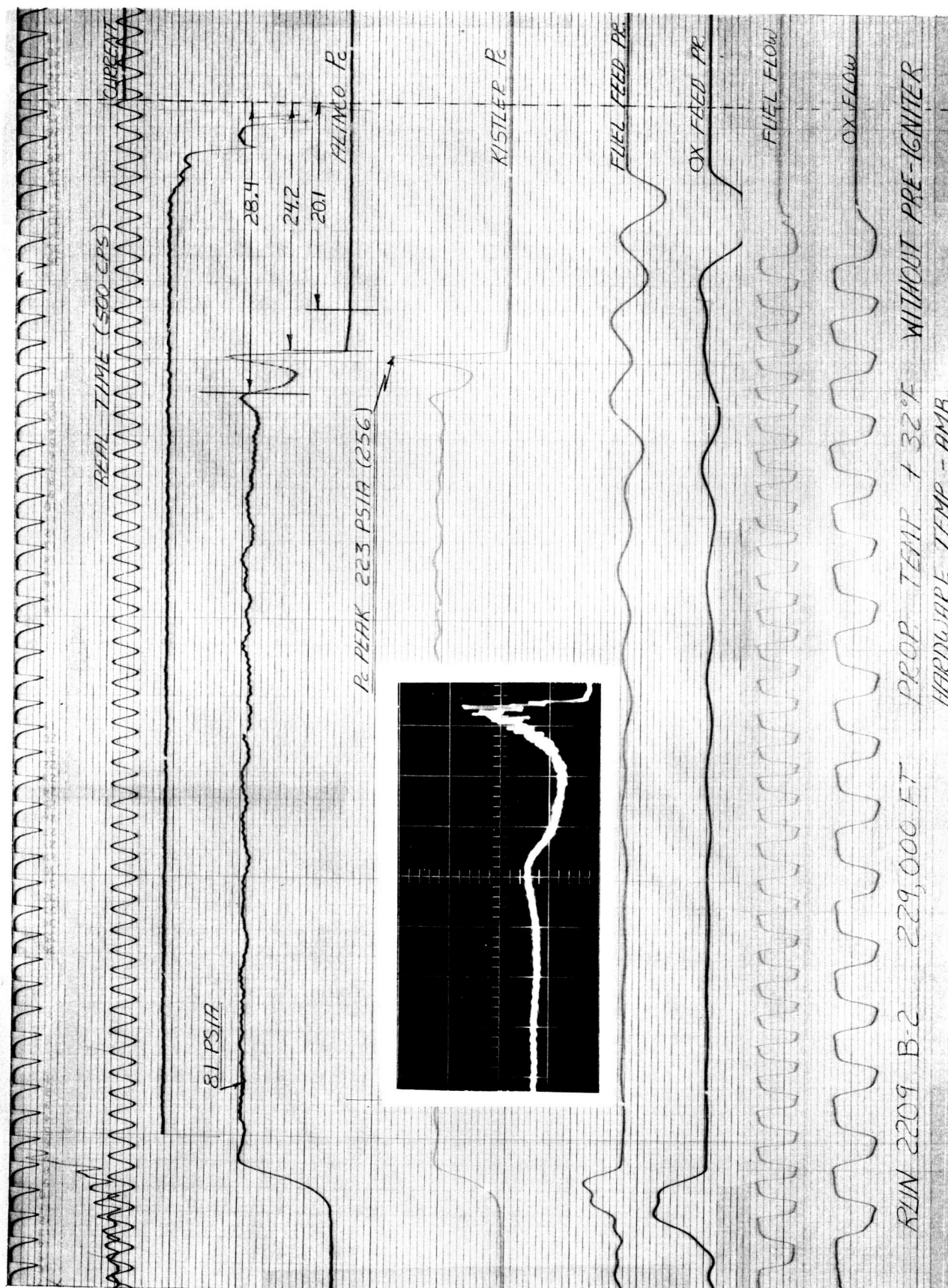


Figure V-123. Run 2208 B-2 229,000 ft



R-196

Figure V-124. Run 2209 B-2 229,000 ft

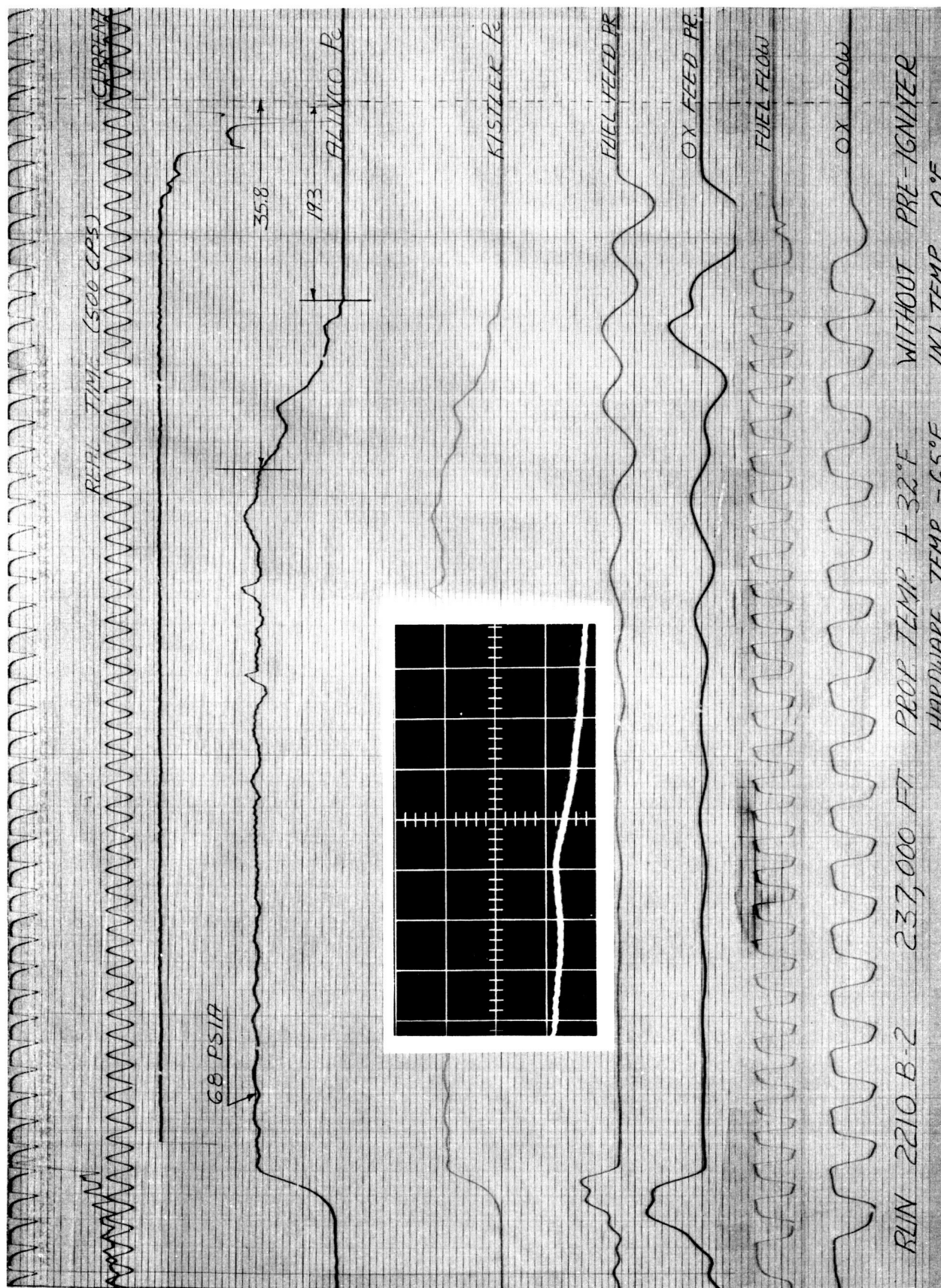


Figure 125. Run 2210 B-2 237,000 ft

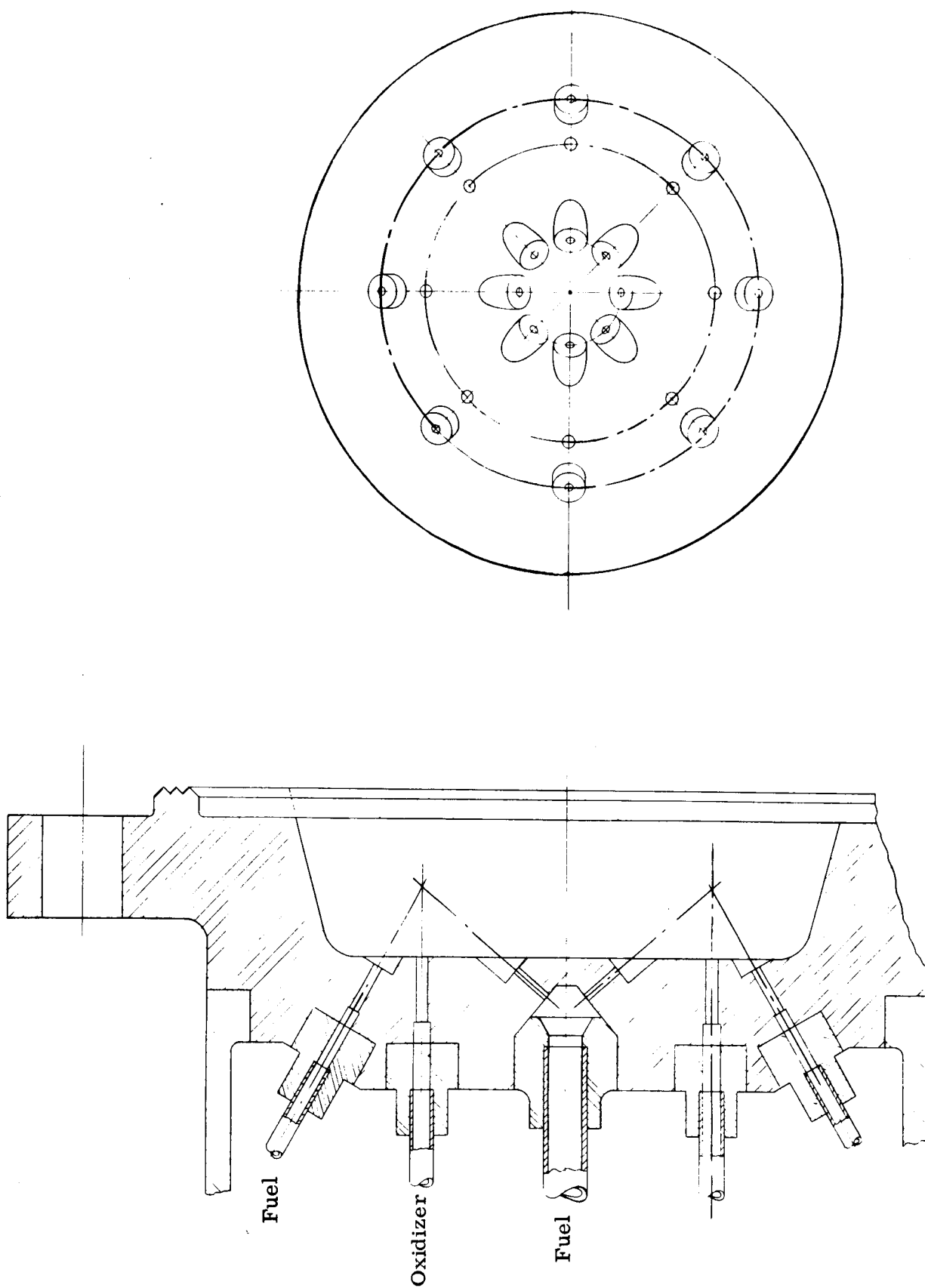


Figure V-126. TU-6 Injector

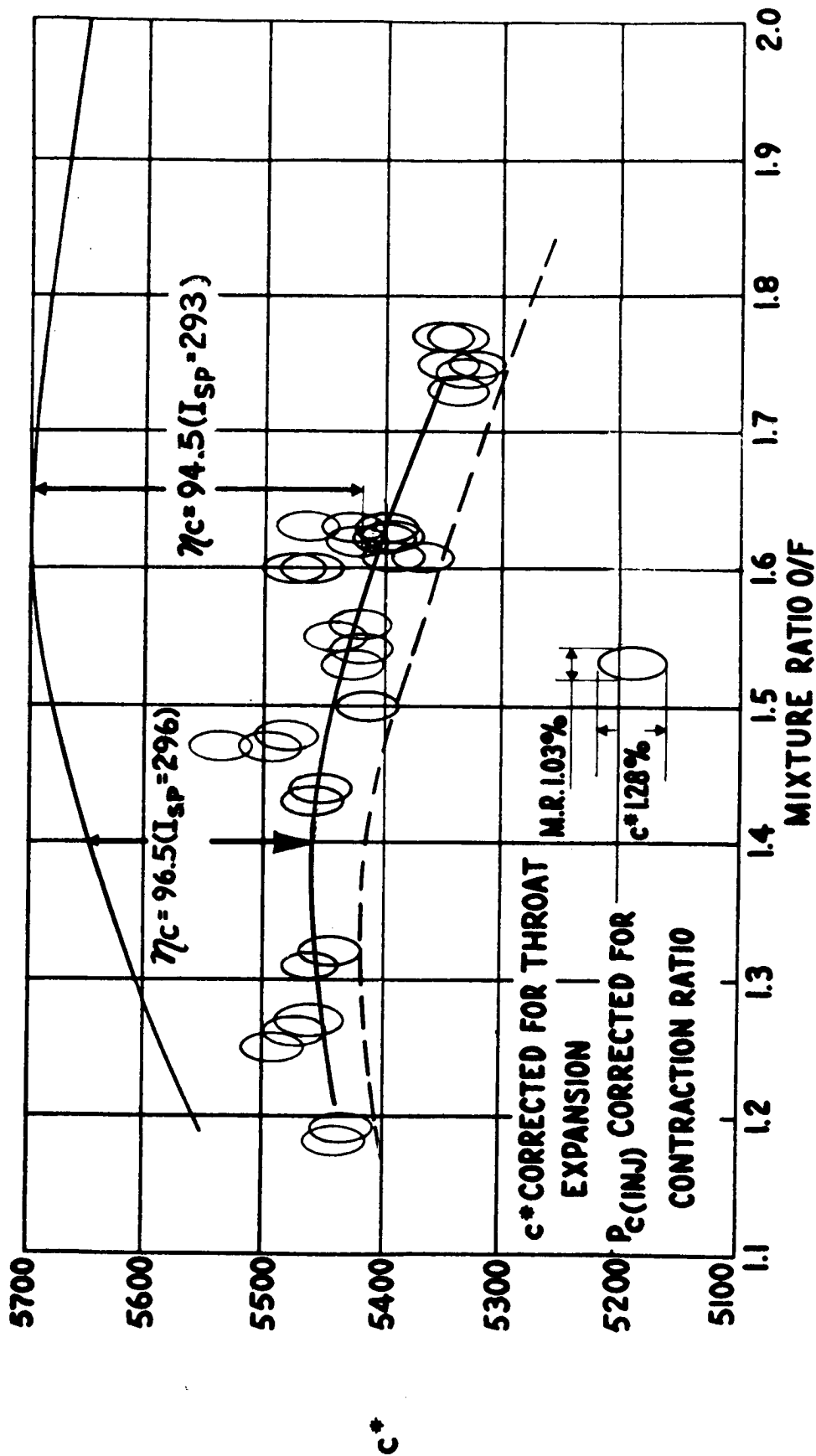


Figure V-127. Injector TU-6A Performance

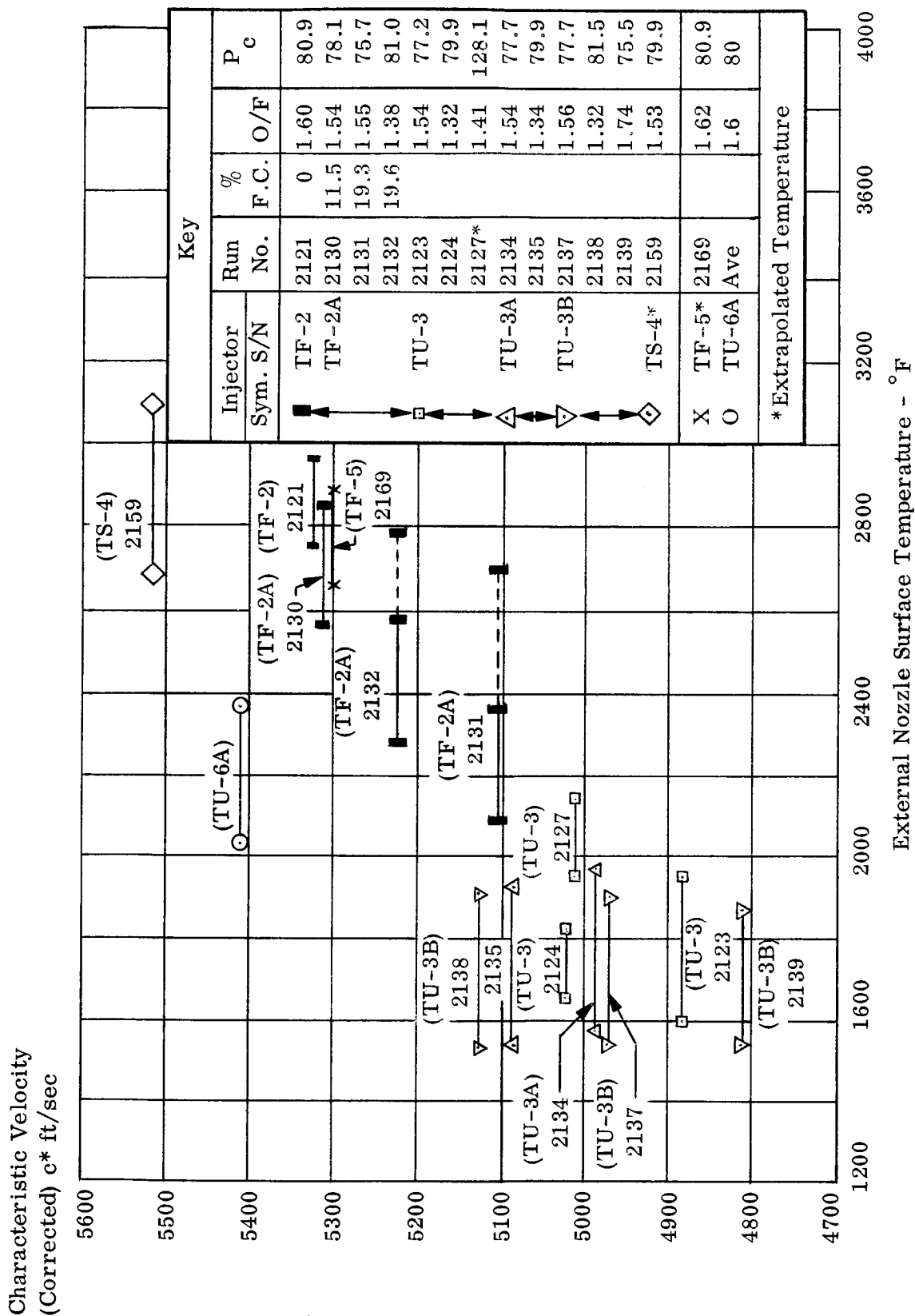


Figure V-128. Model 8374 Nozzle Temperature Data for Various Stabilized Operating Conditions.

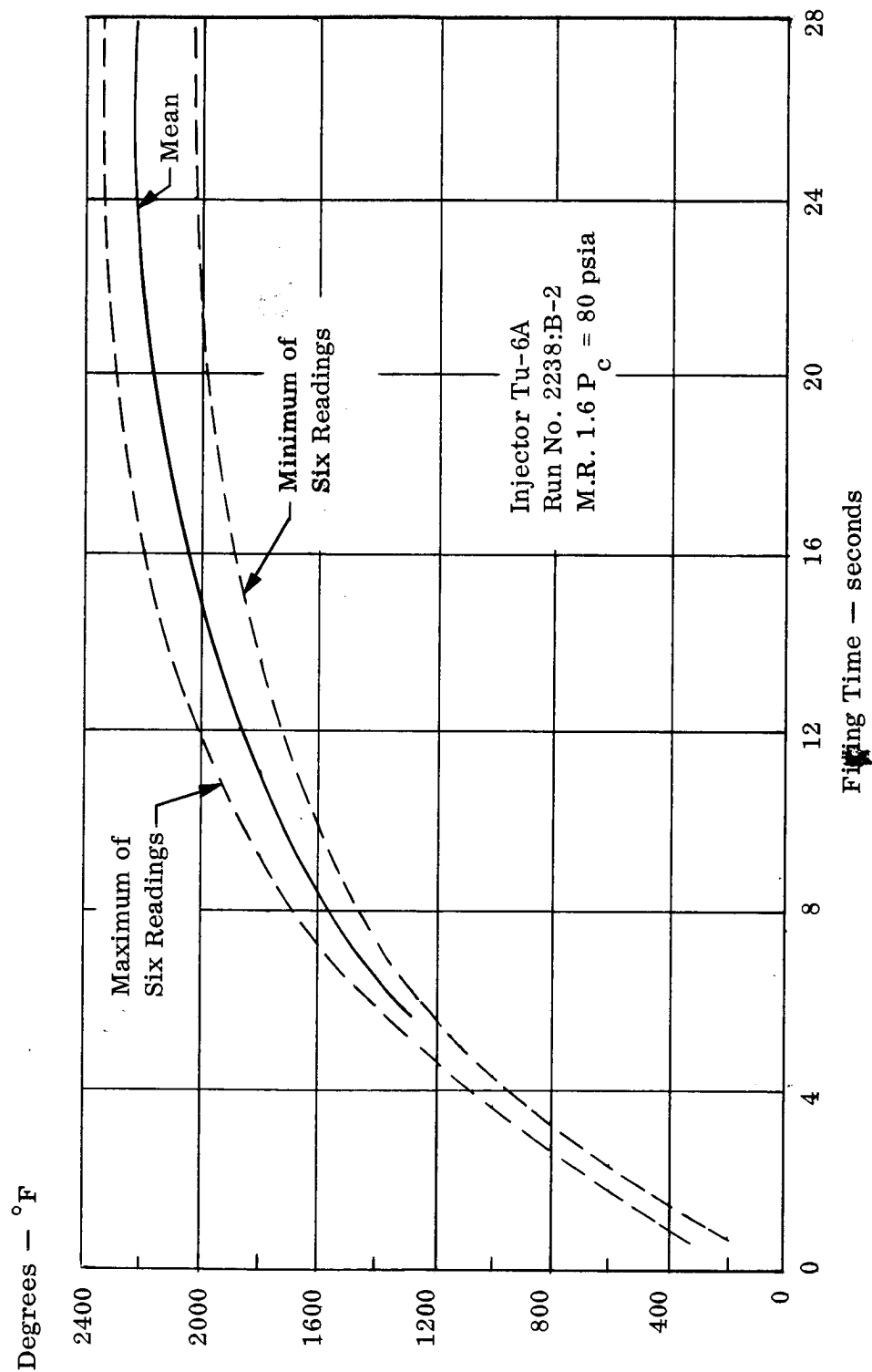


Figure V-129. Throat Surface Temperature

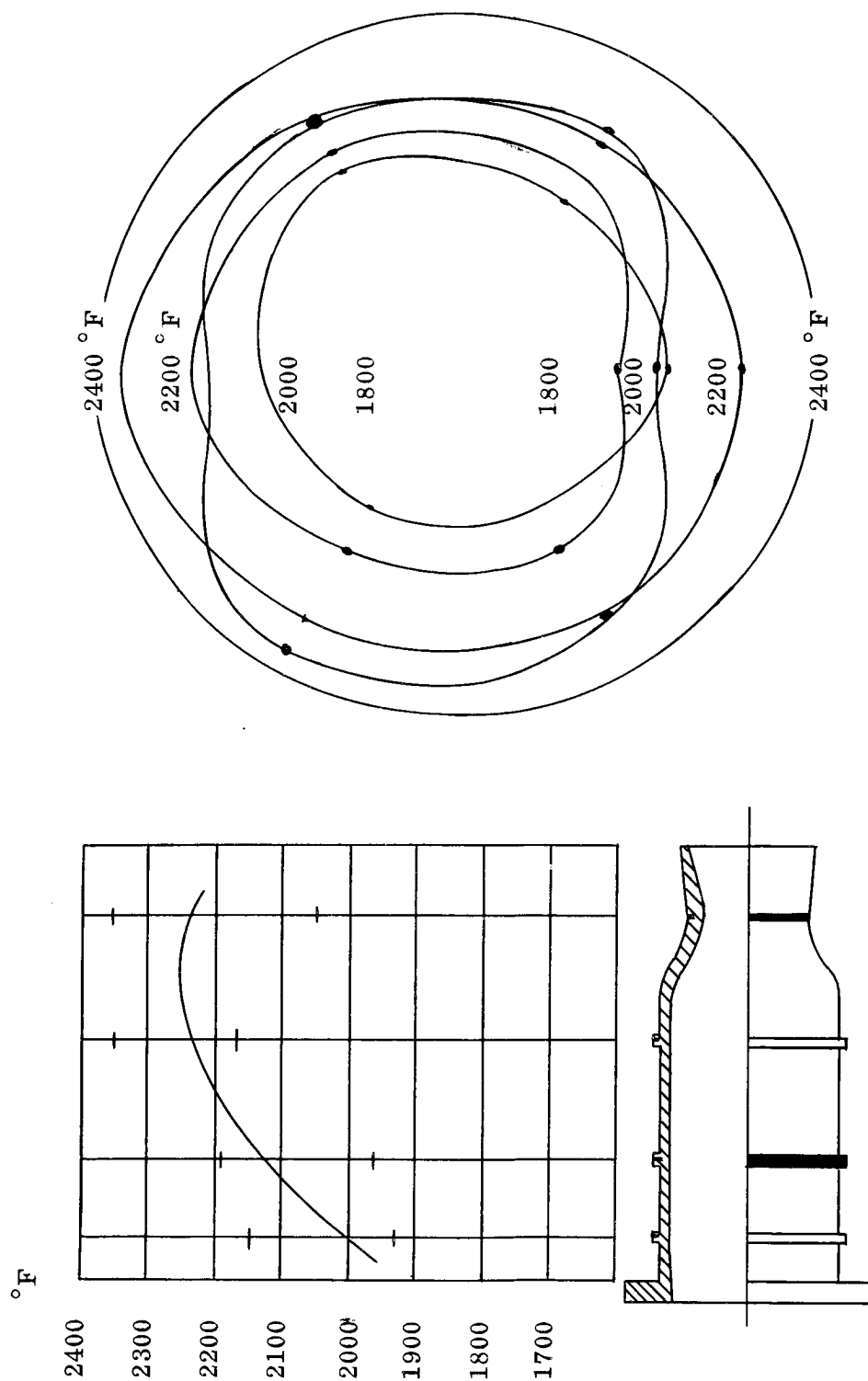
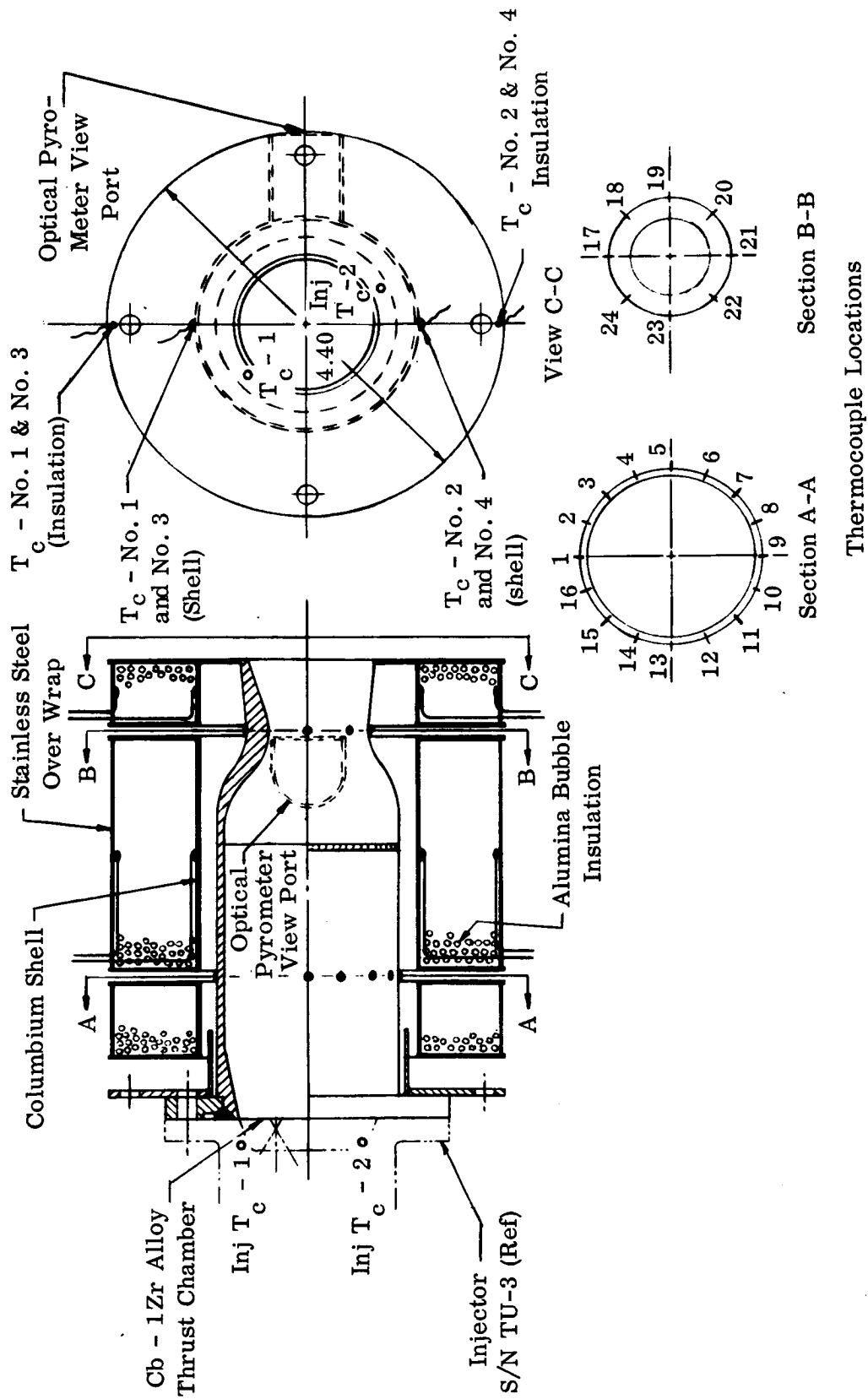
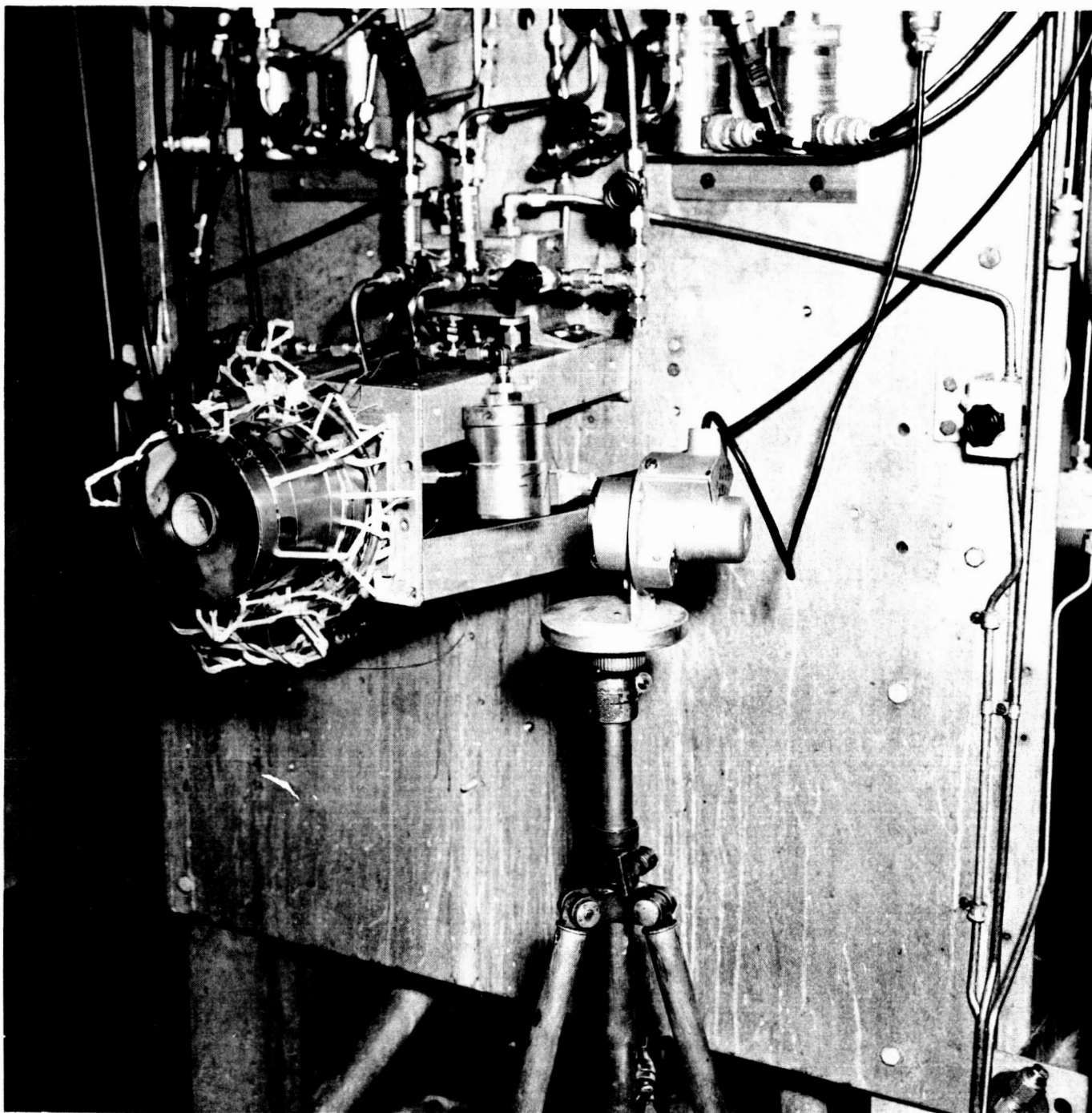


Figure V-130. Free Radiation Wall Temperatures.
Injector TU-6A





204259

Figure V-132. Test Installation - Injector TU-3 Cb Chamber Buried



Adiabatic Wall Thrust Chamber Test
Run No. 2143 Injector TU-3

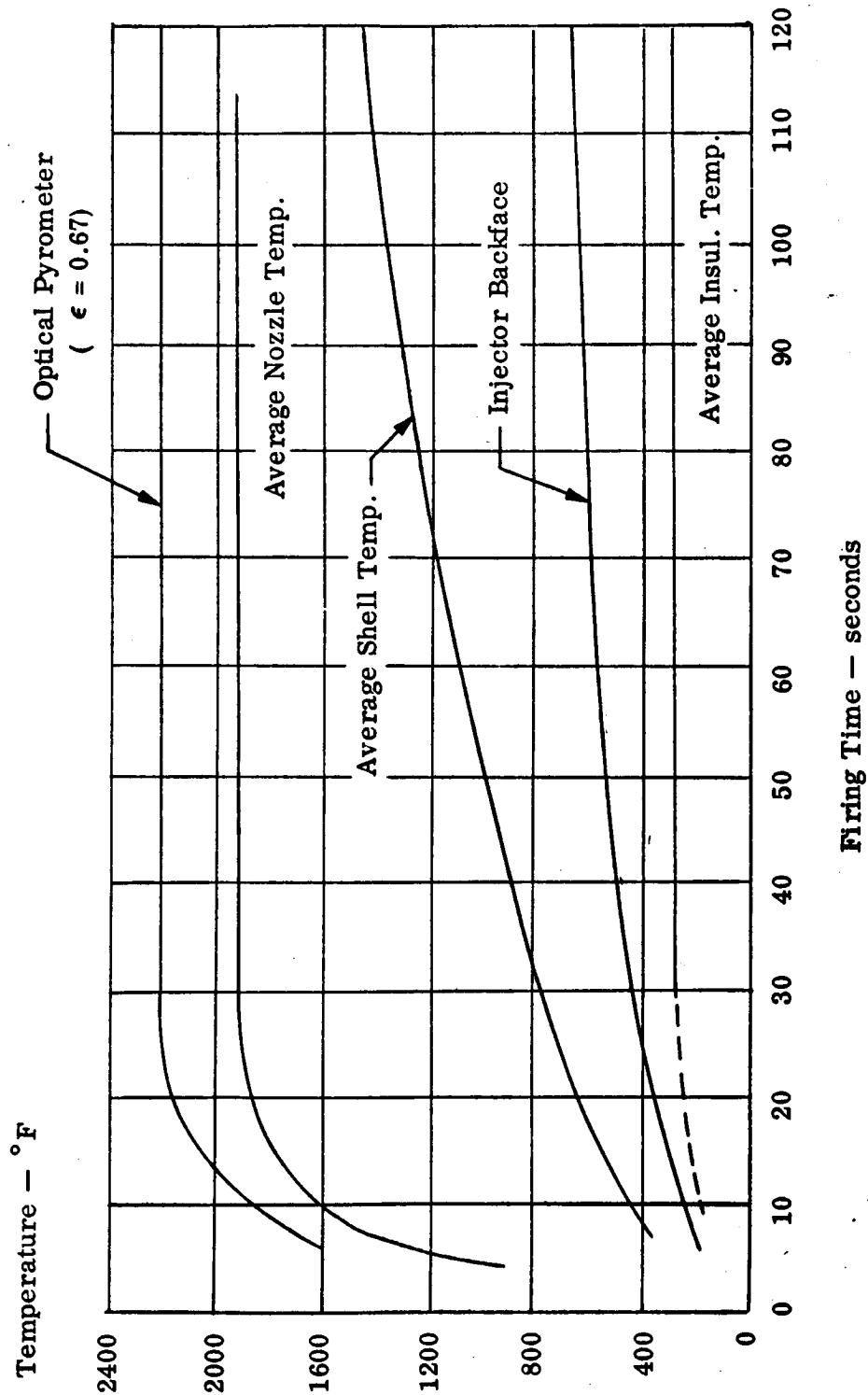


Figure V-133. Model 8374 Average Temperature Profiles

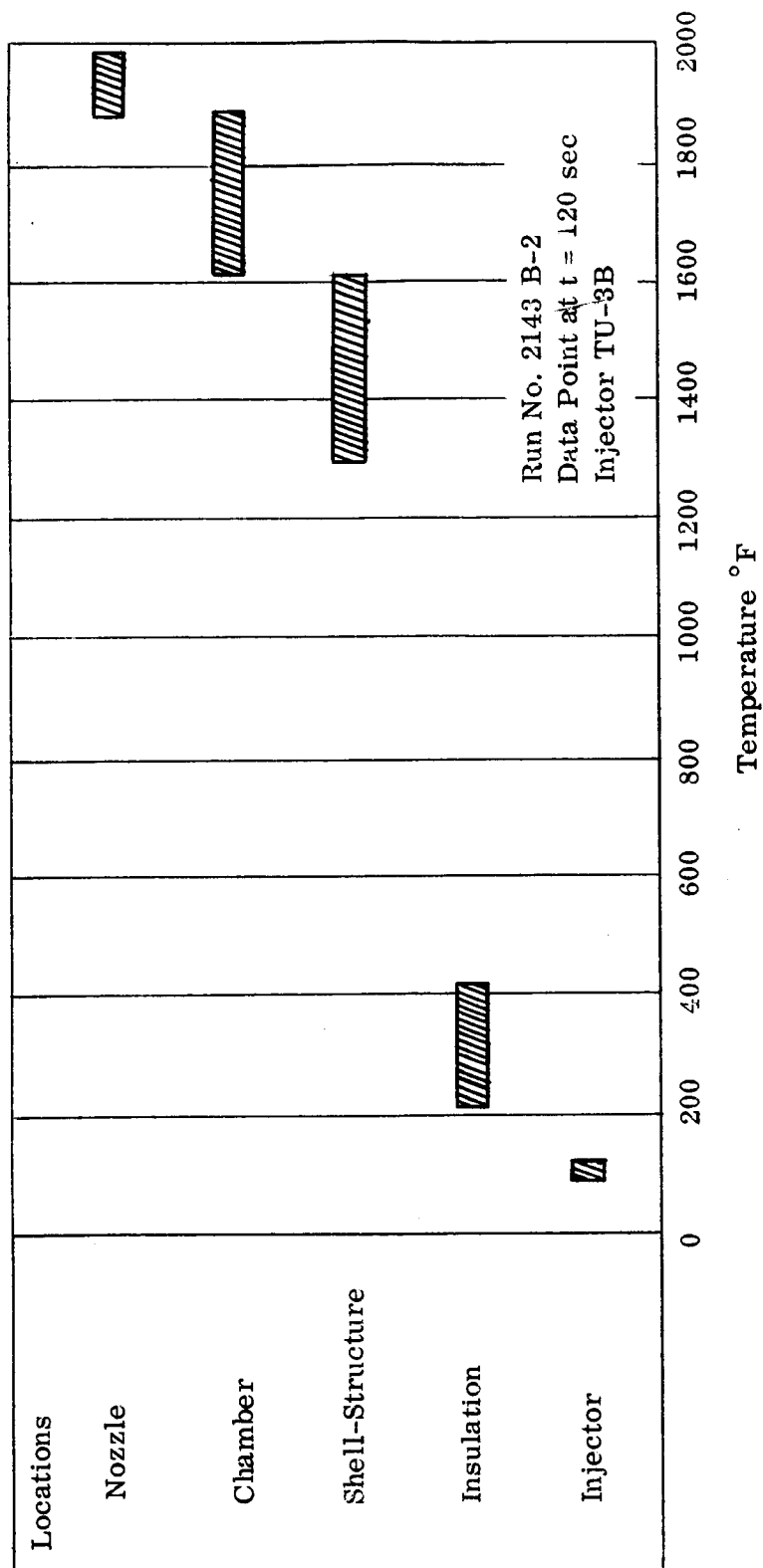


Figure V-134. Temperature Range at Various Thrust Chamber Assembly Locations -
Adiabatic Wall Test

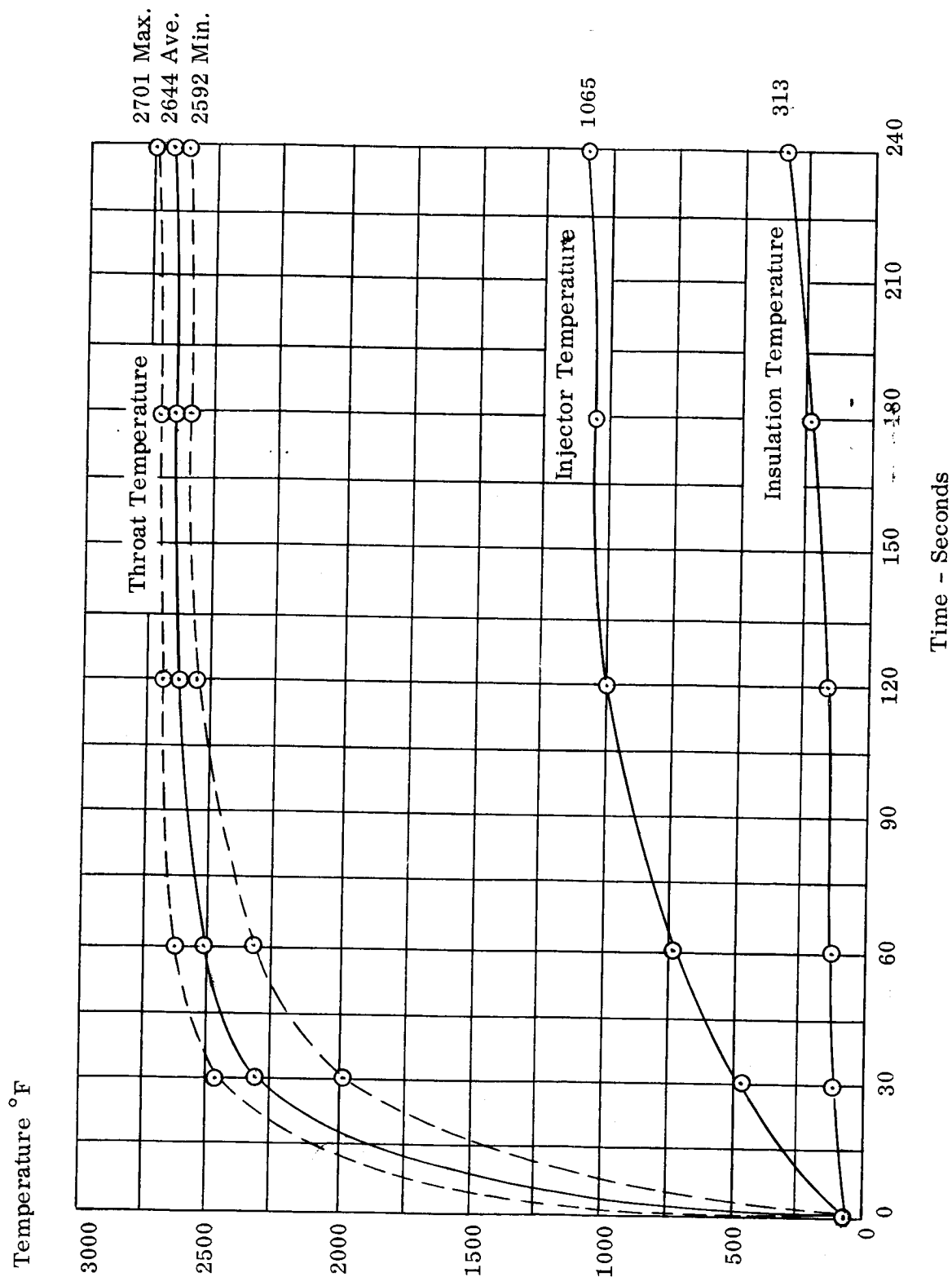


Figure V-135. Injector TU-6A "Buried"

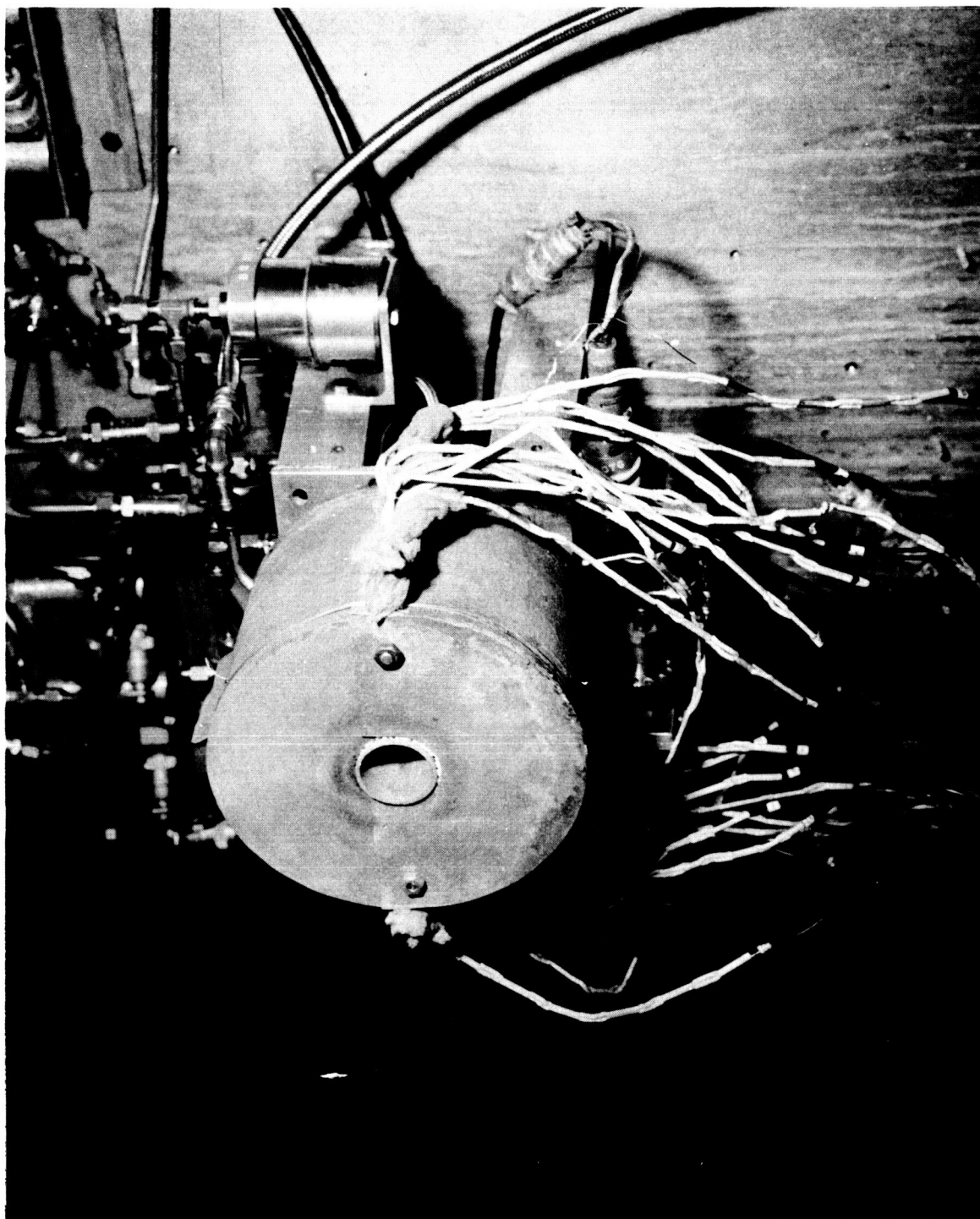


Figure V-133. Buried Test - Cb Chamber - Inj. TU-6



Figure V-137. Buried Test - Cb Chamber - Inj. TU-6 After Removal of Top Half

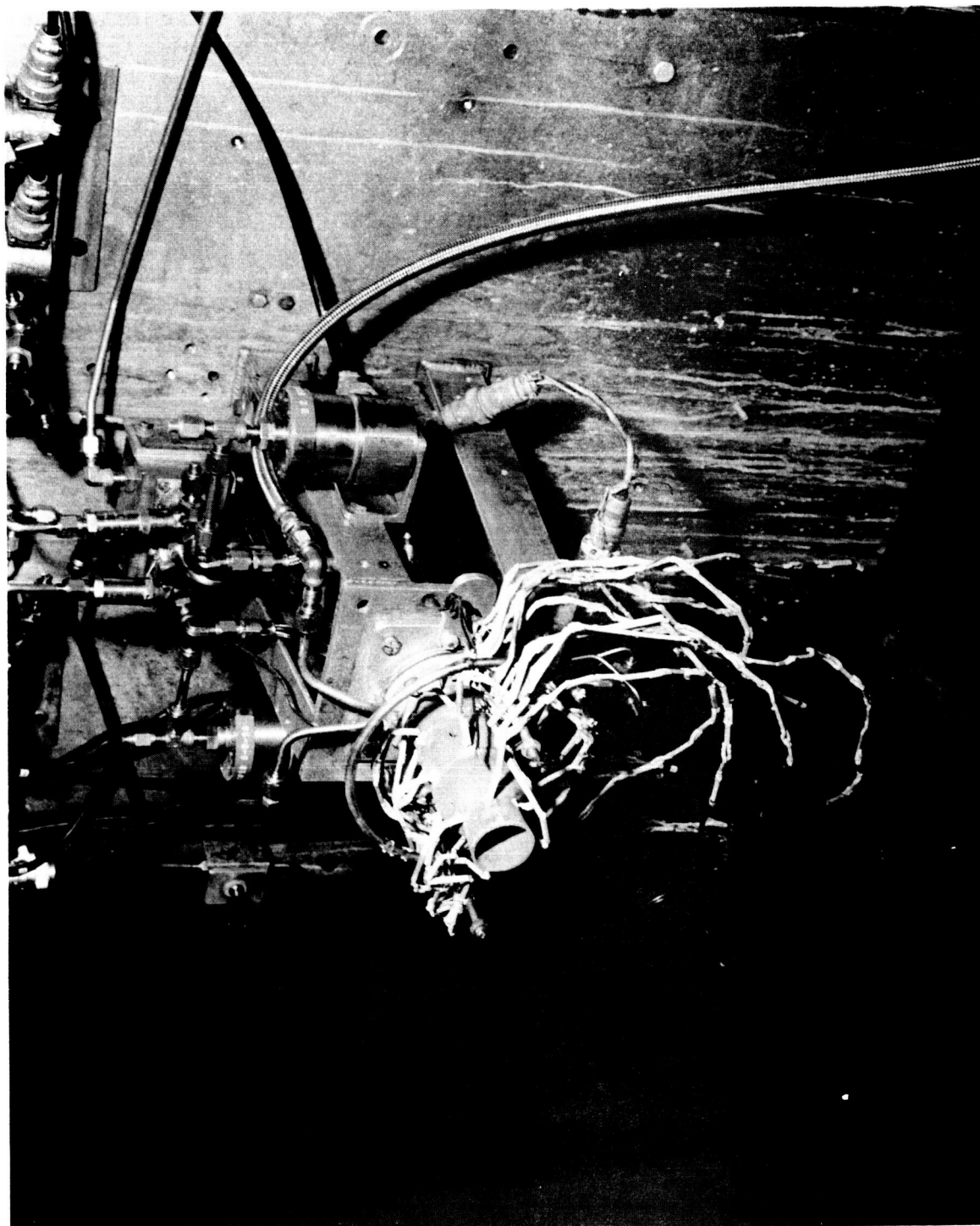


Figure VI-138. Buried Test - Cb Chamber Inj. TU-6 After Removal of Insulation



E. MATERIALS AND COATING EVALUATION

1. Thrust Chamber Materials

Alloys of columbium, tantalum, and molybdenum were considered for this application. The molybdenum alloys were not chosen because the operating temperature, 2800-3500°F, is above their recrystallization temperature. Recrystallization of molybdenum raises the ductile-to-brittle transition temperature, which is an undesirable condition for cyclic conditions. In addition, its welding characteristics are poor, especially when compared to the columbium and tantalum alloys. Since welding is the main method of joining to be used in the fabrication of this engine, reliability would be questionable with molybdenum welds.

A columbium alloy, Cb 291 (10% Ta, 10% W) is desirable for temperatures up to 3000°F. Short-time creep tests at Bell Aerosystems have shown that, based on a strength-to-weight ratio, the Cb 291 alloy has an advantage over Ta-10W up to approximately 3100°F. Cb 291 was chosen instead of other columbium alloys for the same reason. Above 2500°F, other high strength alloys such as Cb 752, have lower short-time creep properties.

2. State-of-the-Art Coatings

The coating which can be utilized for Cb 291 is a modified silicide type developed by GT&E. This coating system has been tested in actual engine firings at Bell Aerosystems in two engines at 2800-2900°F for durations of 15 minutes under various mixture ratio conditions and under a classified Air Force contract to temperatures in excess of 3200°F for short periods.

The expected life under actual service conditions can only be determined by actual firings, but a degree of success can be estimated. Torch tests on silicide coated Cb alloys show that the life increases at a parabolic rate as the difference between the test temperature and the melting point of the coating increases.

<u>Temperature °F</u>	<u>Approximate Hours to Failure</u>
3200	1
3100	8
2900	50
2700	80
2500	100

These values are based on tests conducted at GT&E. Based on the firing experience at Bell, which has yet to show a failure with this coating on radiation cooled chambers at 2800°F, and torch tests which show one hour to failure at 3200°F, it is reasonable to expect at least one hour service life at 2800-3000°F.

The stability of the silicide coating under vacuum conditions has been studied and no problem is foreseen. The coating on the combustion side of the chamber is at temperature under reduced pressures only upon shutdown. The time at which the temperature is in excess of 2500°F is approximately 20 seconds after even steady state firing. Moreover, pulsing will not result in temperatures above 2500°F under most duty cycles.

3. Advanced Coatings

Early in the program a survey was made of major coating vendors to investigate advanced state-of-the-art coating systems for possible application to the prototype engine design. The present coating systems provide oxidation resistance to a temperature value of approximately 3000°F for long term usage. Coatings capable of withstanding temperatures in excess of 3500°F would assure a thermal margin for earth storable propellants and possible application to higher energy propellants.

Initially, no new coating processes appeared attractive for direct application on refractory metal chambers.

a. Hafnium-Tantalum Clad Coating (IITRI)

During the second quarter of this program, IIT Research Institute, Chicago, Illinois was contacted in regard to the hafnium-tantalum clad alloy material they were evaluating under a NASA contract. This coating system incorporated a hafnium-tantalum alloy coating clad to a 90% tantalum - 10% tungsten alloy and showed promise of providing oxidation protection above 3500°F. Techniques had been developed by IIT Research Institute to fabricate hafnium-tantalum clad nozzle inserts by hot spinning. An order was placed for one chamber for fire test and evaluation. This chamber incorporated a hafnium-tantalum coating clad to both the external and internal surfaces of the 90% tantalum - 10% tungsten substrate.

Since the chamber was spun to an existing set of vendor tools, some adaptation was necessary to permit one of the existing test injectors to be assembled to the experimental thrust chamber. As a consequence, a water-cooled adapter section was fabricated to make the transition between the injector and the hafnium-tantalum clad chamber. The drawing in Figure V-139 illustrates the manner in which this transition was accomplished. In addition, this figure also indicates the type and location of the thermocouples used during the subsequent test firings. The photograph in Figure V-140 shows the assembled thrust chamber on the test stand prior to firing.

The injector selected for use with the hafnium-tantalum clad chamber was serial number TF-2 without film cooling flow. An adequate test history had been established for this injector during previous test firings in the water-cooled and columbium thrust chambers. In addition, the injector had been fired without film cooling flow in the columbium chamber for a sufficient duration (including a 30-second test) to achieve thermal equilibrium conditions without damage to the chamber coating. Steady-state temperatures for this run were in the order of 2800 to 3000°F. For these reasons, TF-2 injector was selected to provide a relatively uniform high temperature combustion atmosphere for the hafnium-tantalum clad chamber.

Three firings were made with the chamber in a free radiating condition. The initial firing was a 5-second duration performance check to establish mixture ratio and total flow conditions. The throat area on the test chamber was slightly smaller than that on the columbium (80 psia chamber pressure) thrust chamber. It was decided to maintain the rated total flow of the injector (at a 1.6 mixture ratio) and accept a somewhat higher chamber pressure in the hafnium-tantalum alloy coated chamber. In this manner, the atomization and mixing of the TF-2 injector would be relatively unchanged assuming that the increased back pressure (approximately 20 psi additional chamber pressure) would have only a secondary effect on these characteristics. It was further reasoned that the injector performance and heating rates would not be drastically altered.

A second firing was made for 10 seconds duration to observe the heating patterns prior to subjecting the chamber to a long duration run. The chamber and coating were closely examined at the end of each of these two test runs. No damage to the coating was observed.

The third and final run was made at a mixture ratio of approximately 1.6 and a chamber pressure of slightly more than 100 psia. The run was terminated at the end of thirty-four seconds, at which time the thrust chamber skin temperatures appeared to have stabilized. Examination of the nozzle throat following this run indicated internal coating damage slightly upstream of the nozzle throat station. Further test plans were terminated. Figure V-141 shows the thrust chamber on the test stand following this test series.

A tabulated summary of the test data for the three hafnium-tantalum clad chamber firings is presented in Table V-17. Temperature profiles for the nozzle and chamber barrel appear in Figures V-142 and V-143. It can be seen that the barrel section of the thrust chamber stabilized between 2800 to 2900°F. Temperatures in the nozzle appear to have stabilized in a range slightly higher. Temperature measurement became more erratic at the throat station due to the apparently higher temperatures; however, the general range of temperatures would seem to be between 2800 and 3150°F. The results from the optical pyrometer are higher, with a maximum temperature of nearly 3300°F indicated. It should be noted that a surface emissivity of 0.825 was assumed in determining optical pyrometer temperature data.

A review was held at IIT Research Institute to discuss the test results on the hafnium-tantalum clad chamber and to determine the cause of damage to the internal coating during this test series. A photograph of the two halved sections revealing the defect areas is shown in Figure V-144.

External damage to the hafnium-tantalum clad was also noted in regions where the thermocouples were resistance welded. The type of reaction encountered between the thermocouples and the chamber surface is shown in Figure V-145.

Various metallographic specimens were prepared to aid in determining the cause of the defects. A photomicrograph of the large defect area is shown in Figure V-146, revealing the oxidation of the base metal Ta-10W and some undermining of the Hf-Ta alloy on the downstream edge of the defect. The effect of the thermocouple reaction on the outside surface of the chamber is also shown. Further analysis of this section showed the thermocouple reaction zone extended into the base metal Ta-10W to a maximum depth of approximately 0.009 inch

TABLE V-17
MODEL 8374
TEST DATA SUMMARY
HAFNIUM-TANTALUM TEST CHAMBER
INJECTOR TF-2, 0% FILM COOLING

Run No.	L.O.R. sec	Data P _T sec	$\dot{W}_f$ lb/sec	$\dot{W}_o$ lb/sec	$\dot{W}_{total}$ lb/sec	O/F Ratio	P _c psia	$c^*_{(1)}$ ft/sec	Water-Cooled Chamber Section		
									$\dot{W}_{H_2O}$ lb/sec	T _{in} °F	T _{out} °F
2163	5.0	2.0 4.0	0.133 0.133	0.207 0.207	0.340 0.340	1.56 1.56	103.5 103.3	5288 5278	0.356	63	108
2164	10.0	2.0 4.0 6.0 8.0 10.0	0.132 0.133 0.133 0.133 0.133	0.210 0.210 0.210 0.209 0.209	0.342 0.343 0.343 0.342 0.342	1.58 1.58 1.57 1.57 1.57	104.6 104.5 104.3 104.1 104.2	5315 5294 5284 5285 5286			
2165	34.0	2.0 4.0 6.0 8.0 10.0 15.0 20.0 25.0 30.0	0.132 0.132 0.133 0.133 0.133 0.133 0.133 0.132 0.132	0.208 0.209 0.208 0.207 0.206 0.205 0.206 0.204 0.205	0.340 0.341 0.341 0.340 0.339 0.337 0.338 0.336 0.337	1.57 1.57 1.57 1.56 1.55 1.54 1.55 1.54 1.54	105.2 104.7 104.1 103.6 103.2 102.9 102.0 100.9 100.3	5371 5325 5303 5287 5279 5294 5233 5206 5156	0.371	48	102
										50	105

(1) c^* not corrected for throat expansion

or less than $1/5$ the thickness of the base metal. The small defect area was also examined, revealing the presence of a loose piece of Hf-Ta alloy with the indication of some cracking in the diffusion zone between the Hf-Ta alloy and the pure Ta barrier. Photomicrographs of this area are shown in Figure V-147.

Although a definite cause of the defects could not be determined at this time it is more likely that they were attributed to either the formation of cracks along the diffusion zone and thereby allowing separation of the Hf-Ta alloy; or by the effect of the thermocouple reaction from the exterior surface of the chamber.

Two additional hafnium-tantalum clad chambers were ordered from IIT Research Institute. During the spinning of each blank, cracking was encountered in the nozzle section resulting in unusable chamber sections. Since minor revisions were made in the fabrication cycle of these two chamber sections, two replacement chamber sections were fabricated following the original procedure. Both replacement chamber sections were spun satisfactorily and were received at Bell on August 26, 1965. Both chambers were preoxidized after the spinning cycle and showed no evidence of undesirable surface checking. One chamber is approximately one quarter inch longer than the other, as shown in Figure V-148. Figure V-149 reveals the finish on the inside surface of the chamber and convergent nozzle sections. The longest chamber will be fire tested; the second chamber will serve as a spare.

The chamber was assembled to the aluminum water-cooled adapter section and fire tested, using the 8-element, balanced injector TF-2A. This same adapter section and injector was used during the evaluation of the first hafnium-tantalum clad chamber.

Five firings were made with the chamber in a free radiating condition at sea level conditions. No thermocouples were used due to the possibility of contamination, but one optical pyrometer was positioned to measure the surface temperature at the throat station (at 3:00). The test series consisted of the following firings: two 3-second tests, one 10-second test, one 30-second test, and one 36-second test. A tabulation of the test data is given in Table V-18. The chamber was visually inspected following each firing with very slight throat erosion noted after the 30-second test. The last test was shut down at 36 seconds due to indications of

TABLE V-18
TEST SUMMARY
Hafnium-Tantalum Clad Chamber (IITRI)
Radiation Cooled
Injector TF-2A and Water-Cooled Adapter

Run No.	Run Duration (sec)	Data Point (sec)	Mix Ratio	Pc Test (psia)	Total Flow (lb/sec)	c* Test (ft/sec)	Throat Temperature (one optical Pyrometer) (° F)
2287	3	3	1.71	(note "A")	0.359	(note "A")	-
2288	3	3	1.70	93.1	0.311	5310	-
2289	10	5	1.58	98.5	0.330	5470	-
		9.9	1.57	97.9	0.330	5425	-
2290	30	5	1.56	98.2	0.330	5450	-
		15	1.56	96.9	0.330	5380	-
		29.3	1.56	96.3	0.330	5340	(note "B")
2291	36	5	1.57	96.3	0.331	5330	-
		15	1.57	95.5	0.331	5285	-
		35.5	1.57	(note "A")	0.331	(note "A")	(note "B")

(note "A") = Pc Data Unreliable

(note "B") = Temperature data obtained with optical pyrometer
Indicated values below 2300° F
Assumed temperature range between 2800 - 3000° F

throat erosion. Postrun inspection revealed severe erosion at three locations in the throat region with a small "burn through" at the 3:00 position. The condition of the chamber following this test series is shown in Figures V-150 through V-152.

No additional testing of this type of chamber is planned during this program.

b. Hafnium-Tantalum Coating (Slurry Process - GT&E)

The Sylcor Division of GT&E evaluated a slurry-applied hafnium-tantalum coating on a tantalum-10% tungsten substrate and on a columbium substrate. Results of oxy-acetylene torch tests on various test coupons showed very satisfactory results with good adherence and high density indicated.

A columbium chamber was fabricated including grooved thermocouple rings coated by GT&E and tested at Bell under Bell sponsorship.

Two test series were performed on this chamber at sea level conditions. The first series was conducted using the unbalanced injector TU-6A with the chamber insulated with a columbium shell and alumina bubbles. The expected chamber temperature under these conditions is approximately 2700°F. This series consisted of one 5-second, one 10-second, one 30-second, and one 240-second test accumulating a total of 285 seconds. One optical pyrometer was used to measure the throat station temperature at the 3:00 position. No thermocouples were attached to the chamber but eight thermocouples were used for monitoring the columbium insulation shell temperature and the insulation, chamber flange, and injector temperature. A summary of the test data from this series is shown in Table V-19. The installation of the chamber in the test cell following this series is given in Figure V-153. Three views of the chamber after removal from the test stand are shown in Figures V-154 through V-156. A postrun inspection revealed no internal erosion. Oxidation of the exterior surface occurred near the convergent nozzle section at the thermocouple rings at the 9:00 position. These rings were provided for the possible installation of thermocouples for future tests. It appears that adherence of the coating was not completely satisfactory at the sharp corners of these rings, resulting in apparent oxidation of the substrate.

TABLE V-19
TEST SUMMARY

Columbium Chamber - Hafnium-Tantalum Coated (Sylcor)
Insulated with Columbium Shell and Alumina Bubbles
Injector TU-6A

Run No.	Run Duration (sec)	Data Point (sec)	Mix Ratio	Pc Test (psia)	Total Flow (lb/sec)	c* Test (ft/sec)	Throat Temperature (one optical Pyrometer) (° F)
2292	5	5	1.60	84.6	0.333	5385	-
2293	10	5	1.58	84.6	0.334	5375	-
2294	30	9.5	1.61	84.3	0.332	5380	2070
		5	1.58	85.5	0.336	5400	-
		15	1.57	84.9	0.335	5365	2460
		29.3	1.59	84.3	0.334	5360	2675
2295	240	5	1.60	85.2	0.333	5430	2350
		15	1.59	84.5	0.332	5390	2600
		30	1.59	84.0	0.332	5360	2595
		100	1.57	84.2	0.333	5360	2535
		239	1.56	83.5	0.331	5350	2530

(note "A") = An emissivity of 0.7 was assumed. (With an emissivity of 0.825 the temperature would be reduced approximately 100° F)

TABLE V-19 (CONT)

Radiation Cooled
Injector TF-2A

Run No.	Run Duration (sec)	Data Point (sec)	Mix Ratio	Pc Test (psia)	Total Flow (lb/sec)	c* Test (ft/sec)	Throat Temperature (one optical Pyrometer) (°F)
2296	5	5	1.56	84.1	0.332	5370	-
2297	10	5	1.57	84.7	0.333	5385	-
		9.5	1.58	83.9	0.334	5325	1700
2298	30	5	1.57	84.7	0.335	5365	-
		15	1.57	83.7	0.335	5300	2400
		29.3	1.58	83.7	0.333	5320	2910
2299	45	5	1.57	84.7	0.333	5395	-
		15	1.57	83.7	0.332	5340	2360
		30	1.57	83.5	0.332	5335	2845
		45	1.57	83.5	0.331	5340	2925
2300	49	5	1.57	85.1	0.332	5420	-
		15	1.57	84.1	0.332	5360	2390
		30	1.57	83.8	0.332	5345	2815
		45	1.57	83.5	0.332	5330	2910

(note "A") = An emissivity of 0.7 was assumed. (With an emissivity of 0.825, the temperature would be decreased approximately 100° F.)

The second test series was conducted under free radiation conditions using injector S/N TF-2A. This allowed visual observation of the exterior surface of the chamber while subjecting the chamber to temperatures near 3000°F. This series consisted of five tests accumulating 139 seconds. The total accumulated time on the chamber is 424 seconds. The duration of each test was as follows: 5, 10, 30, 45, and 49. The chamber was visually inspected after each test, with slight erosion of the throat noted after the 45-second test. The last test was shut down at 49 seconds due to erosion in the throat region. Postrun inspection revealed severe internal erosion at the throat at the 3:00 position, and erosion in the chamber section at the 1:00 and 5:00 positions. External oxidation is extreme at the 1:00 and 5:00 positions also.

Two views of this chamber following the test series are shown in Figures V-157 and V-158. No further testing is planned at this time.

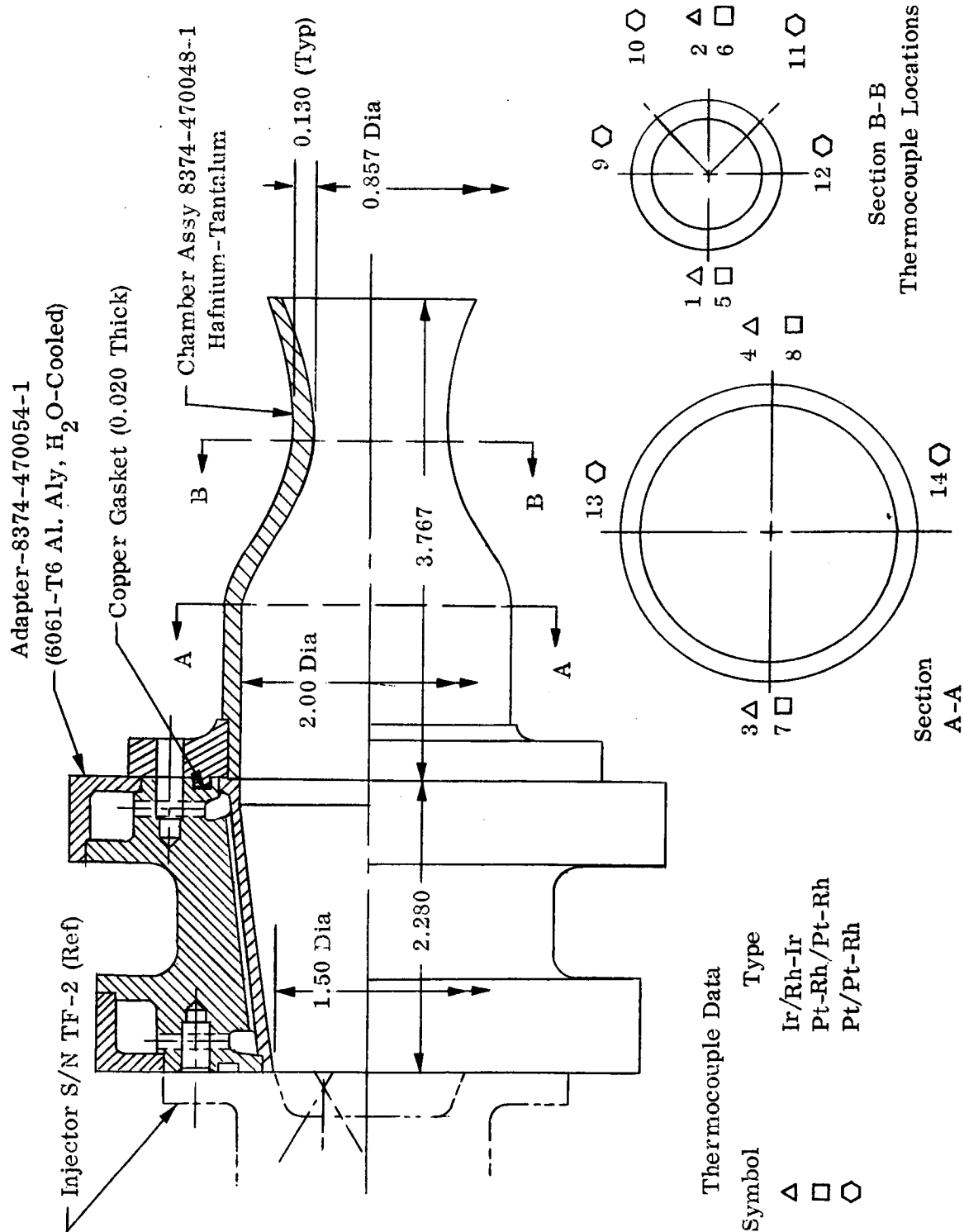


Figure V-139. Hafnium - Tantalum Chamber Assembly

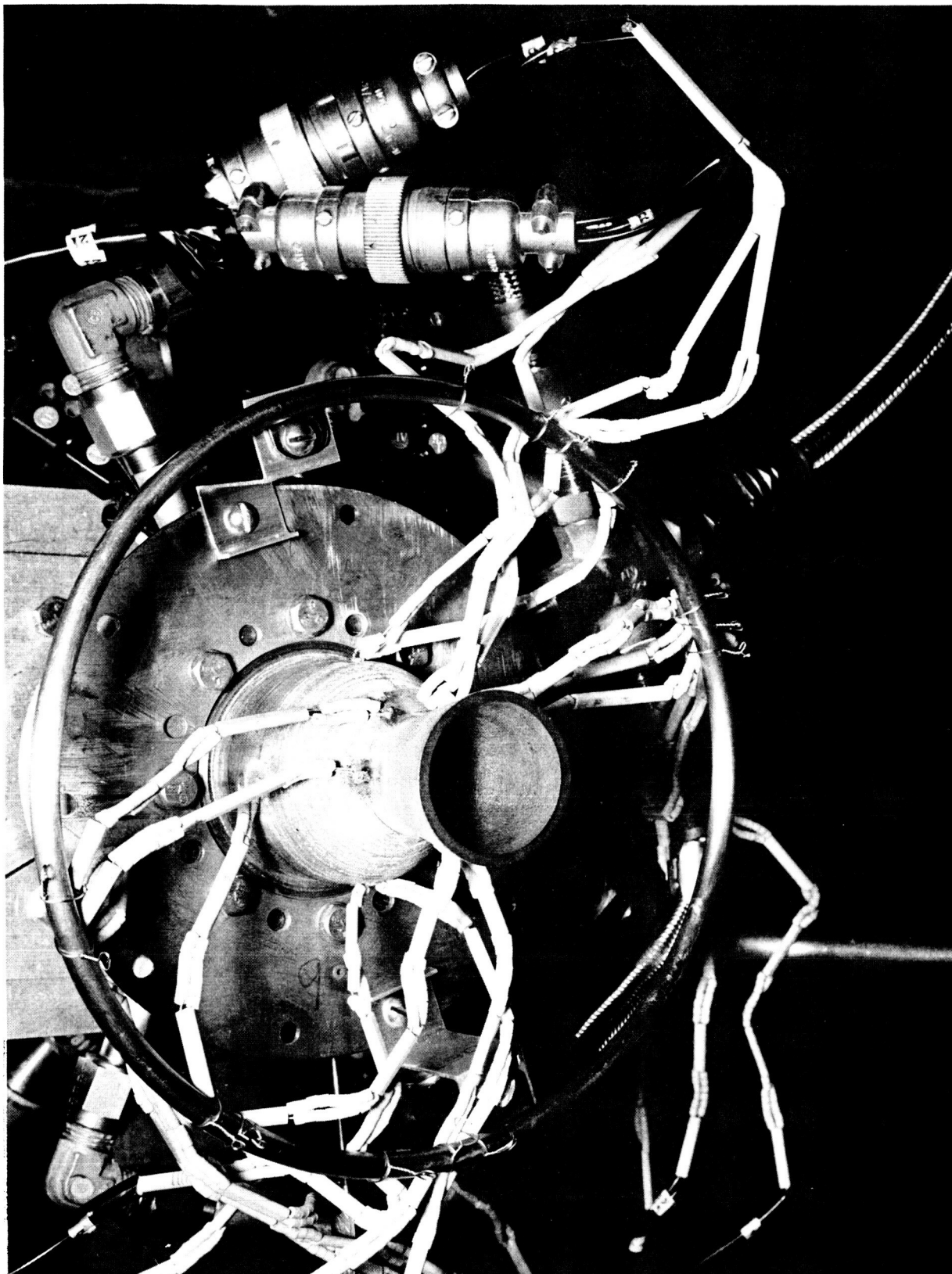


Figure V-140. Hafnium - Tantalum Chamber Test Setup Before Fire Test

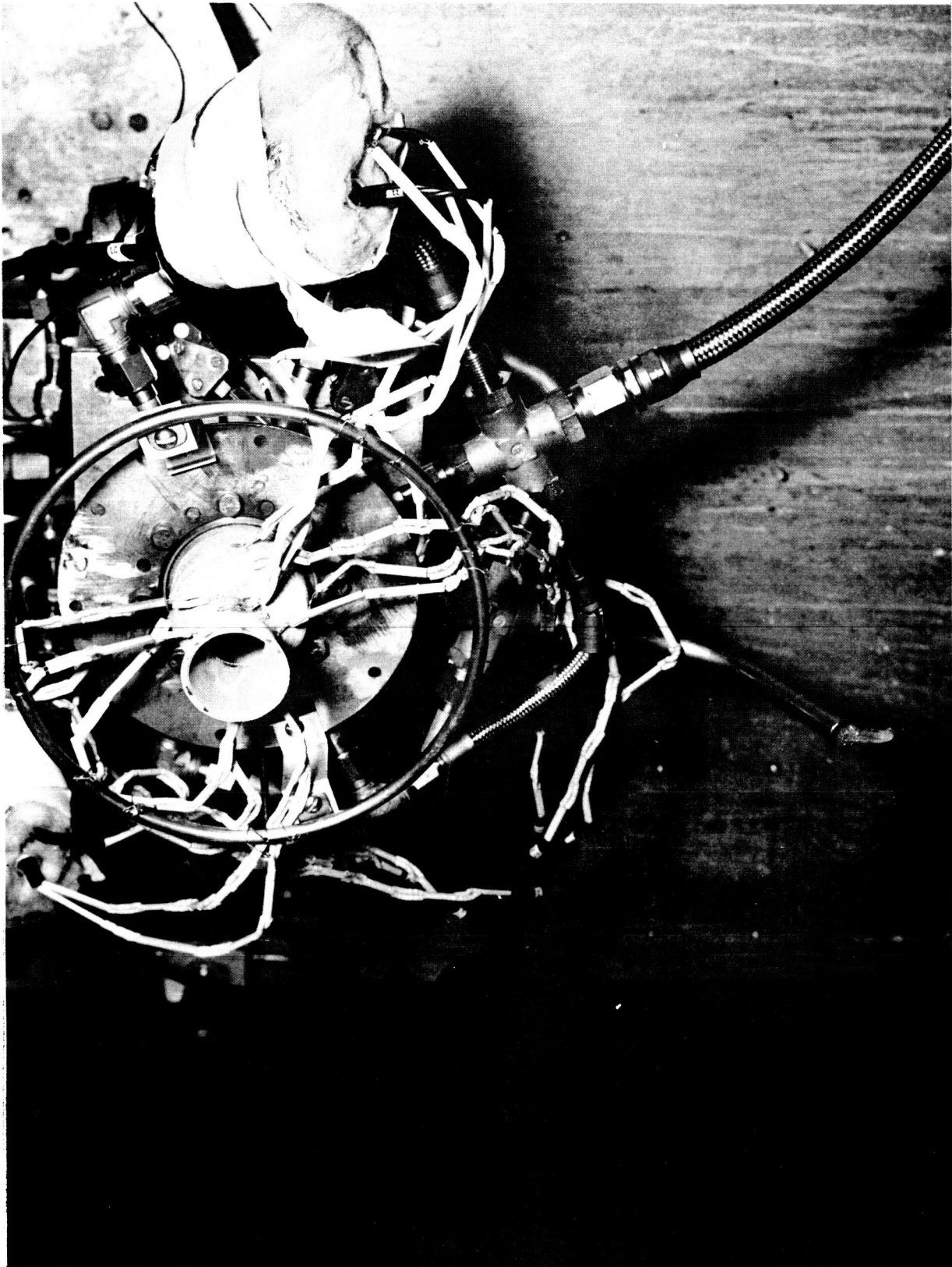


Figure V-141. Hafnium - Pantalum Chamber Test Setup After Fire Test

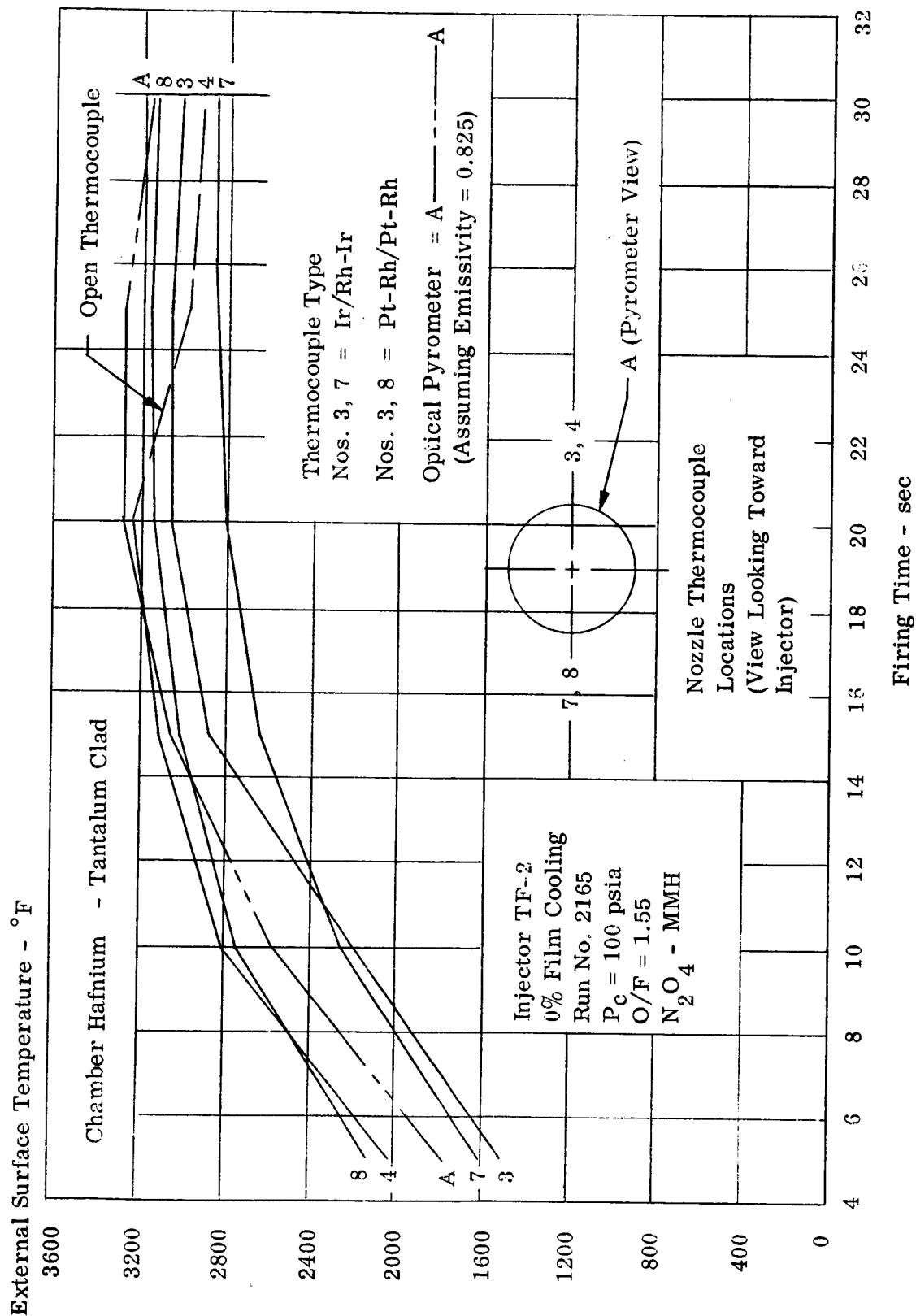


Figure V-142. Nozzle Temperature Versus Firing Time

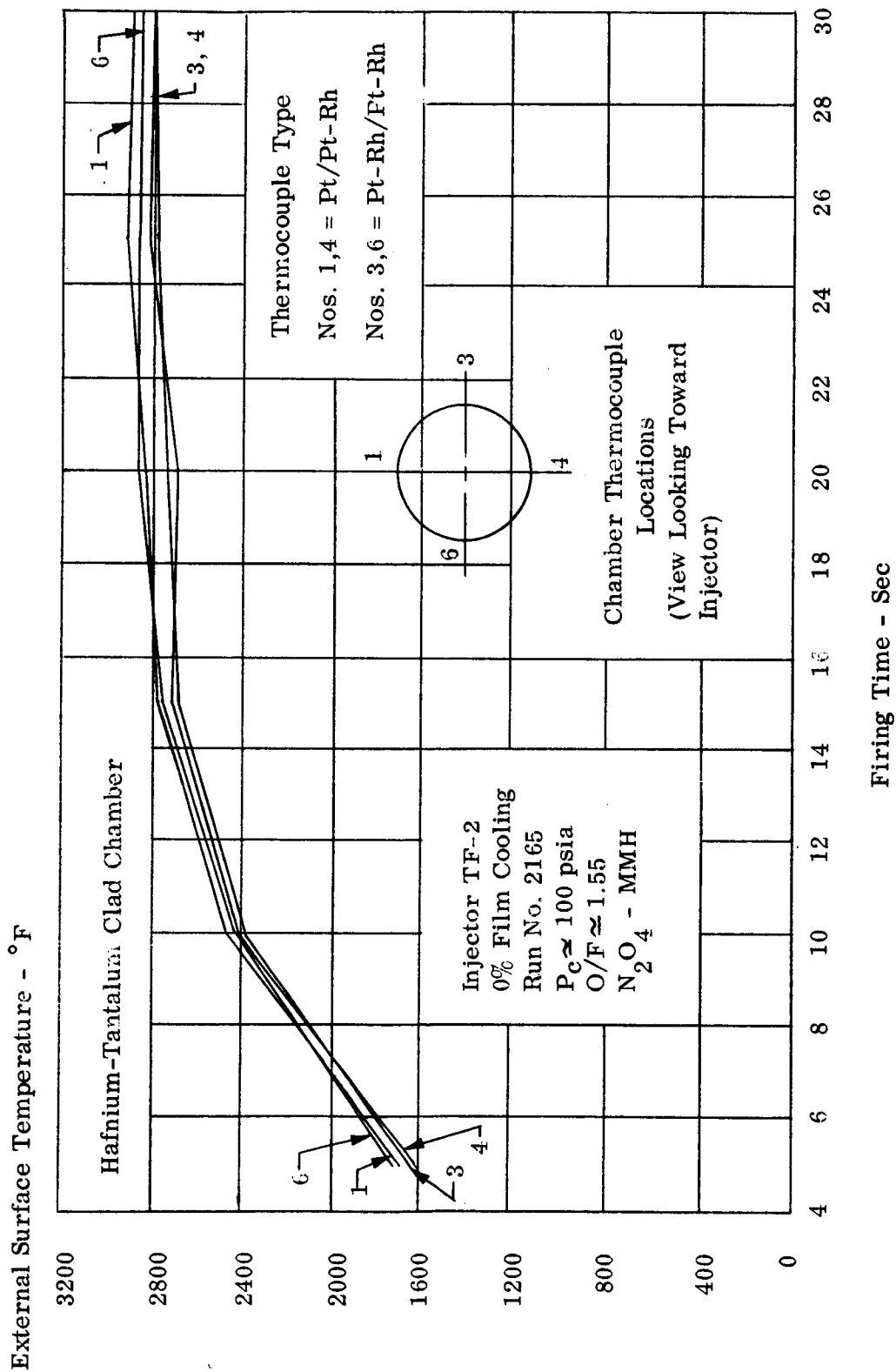


Figure V-143. Chamber Temperature Versus Firing Time

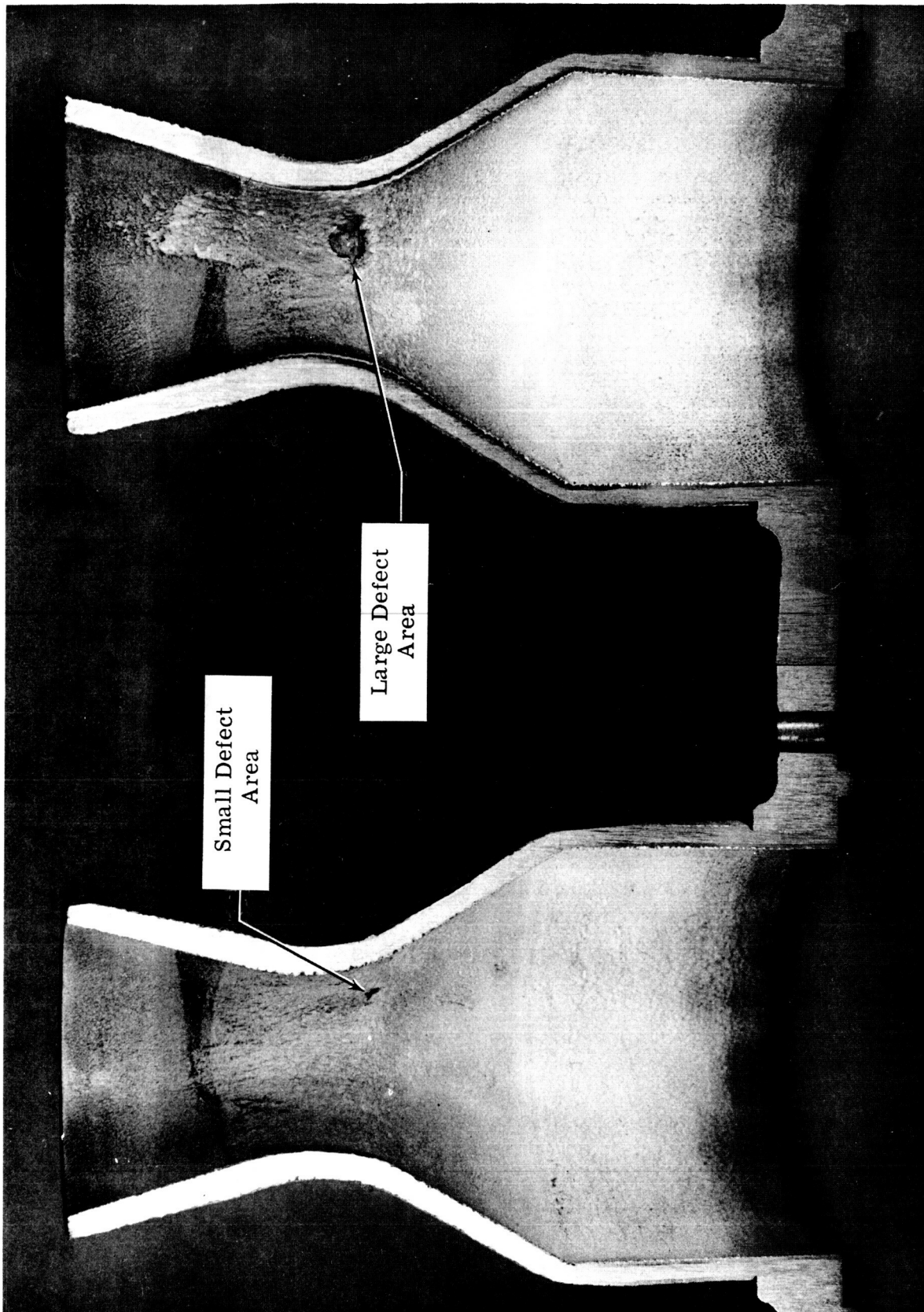


Figure V-144. Hafnium - Tantalum Chamber - Sectioned After Fire Test

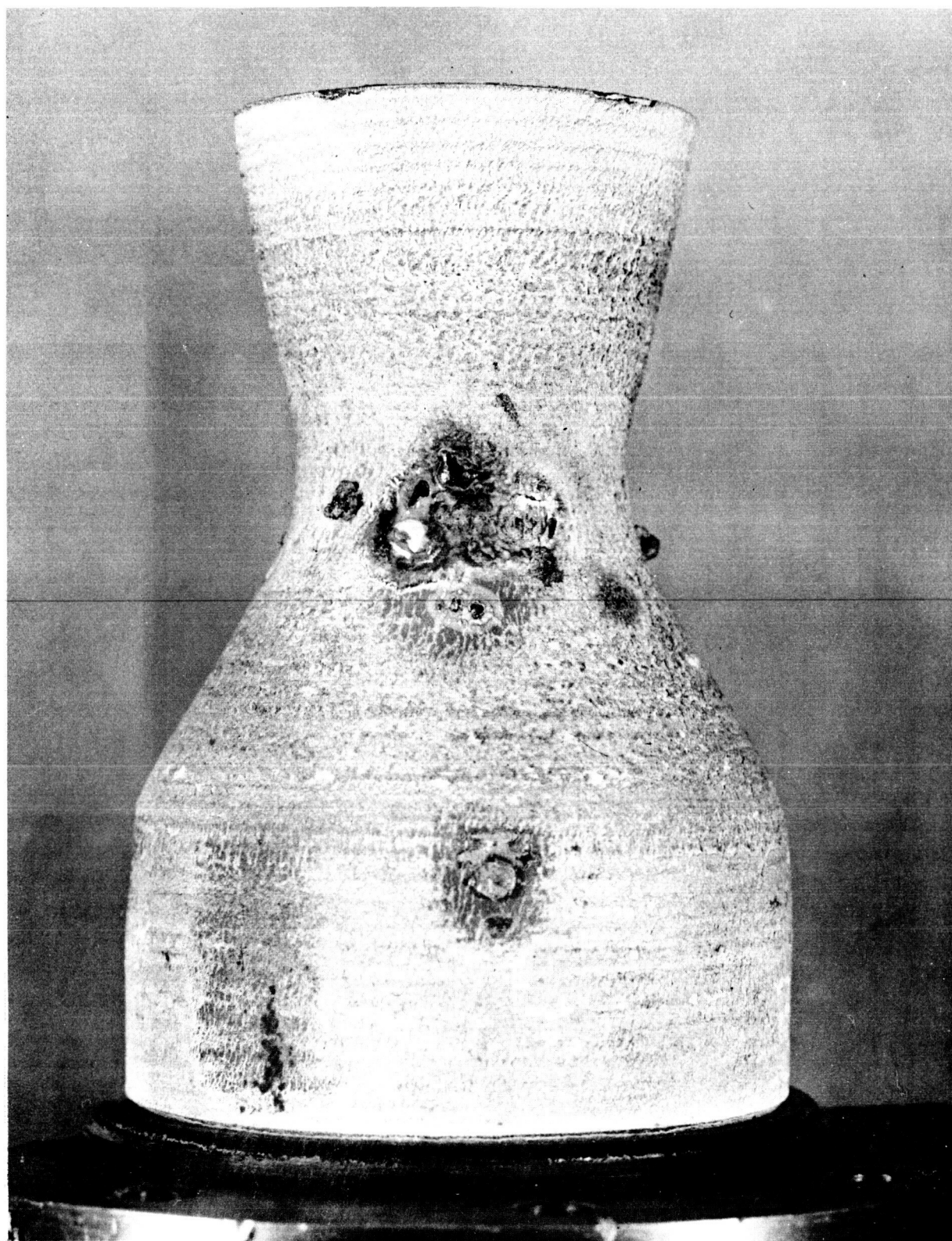


Figure V-145. Hafnium - Tantalum Chamber - Exterior Thermocouple
Reaction After Fire Test

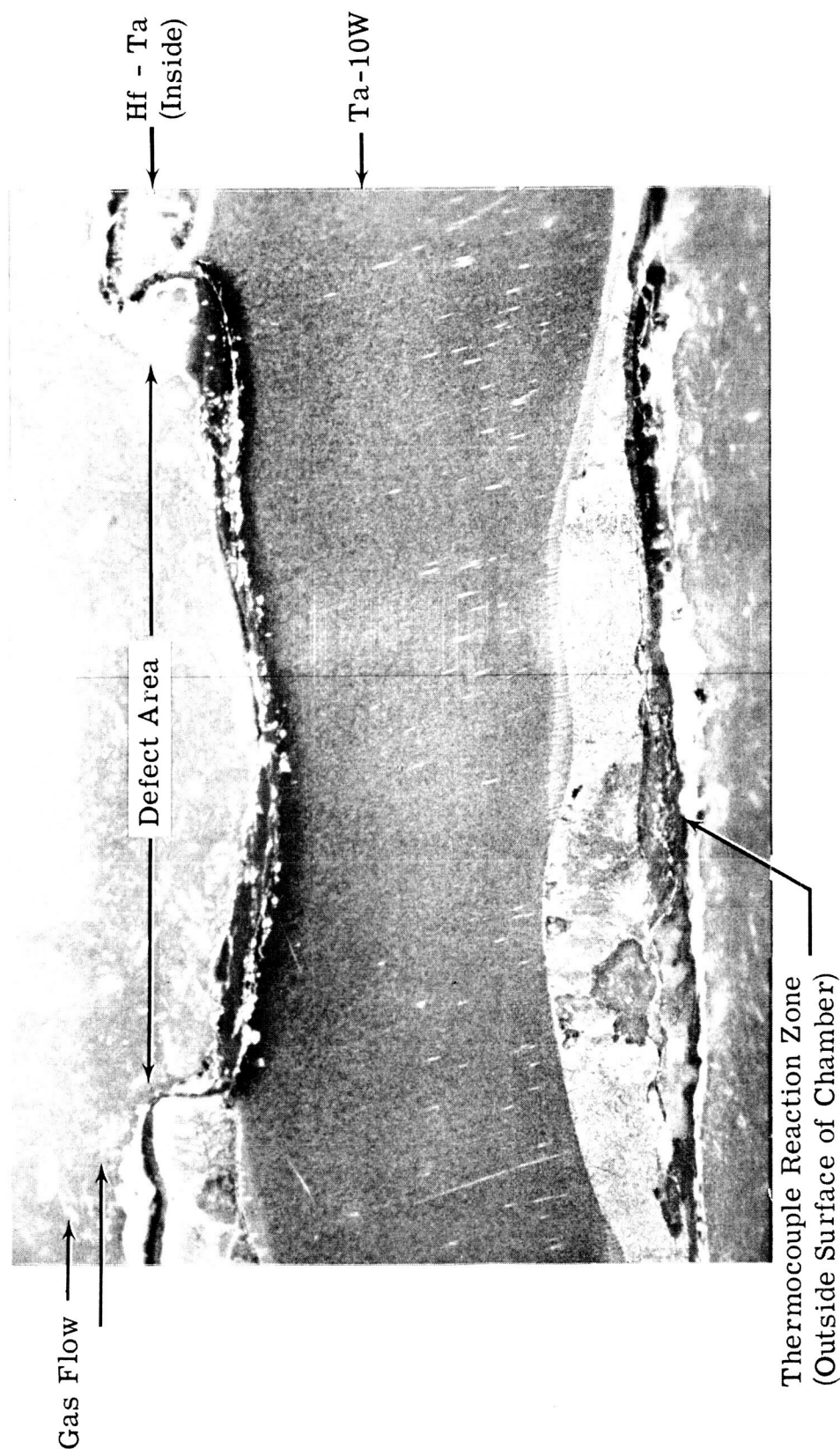
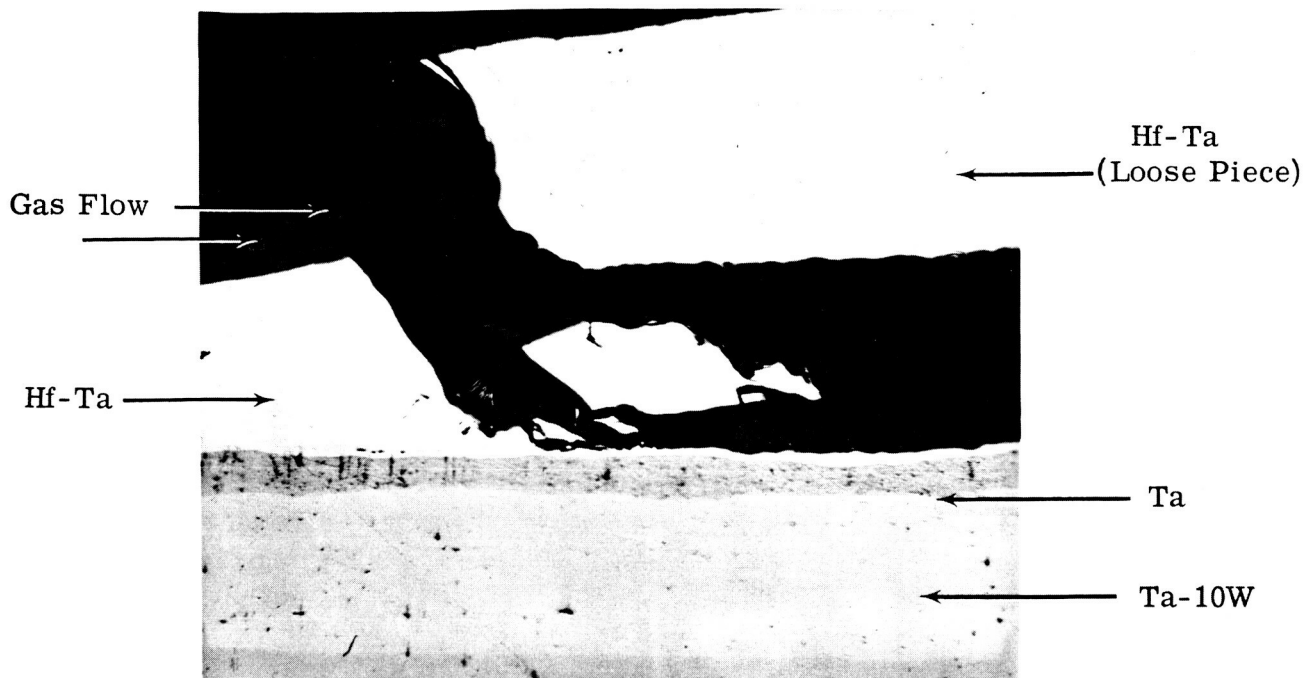
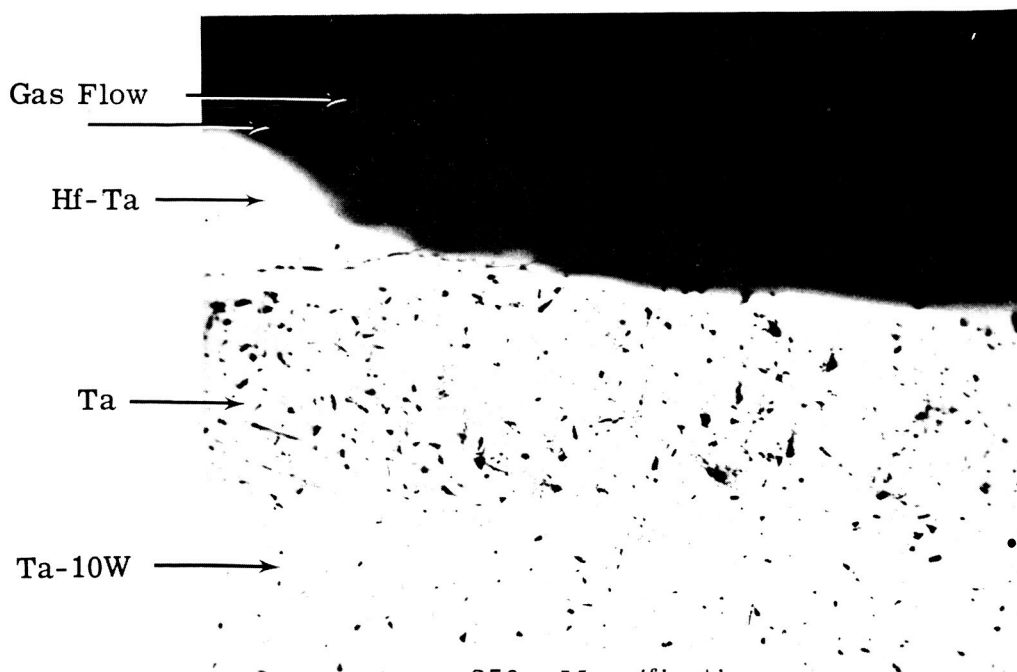


Figure V-146. Photomicrograph of Large Defect Area - 30x Magnification



75 x Magnification
0.010" Above Defect Area



250 x Magnification
At Defect Area

Figure V-147. Small Defect Area - Inside Surface



Figure V-148. Hafnium - Tantalum Chambers After Spinning

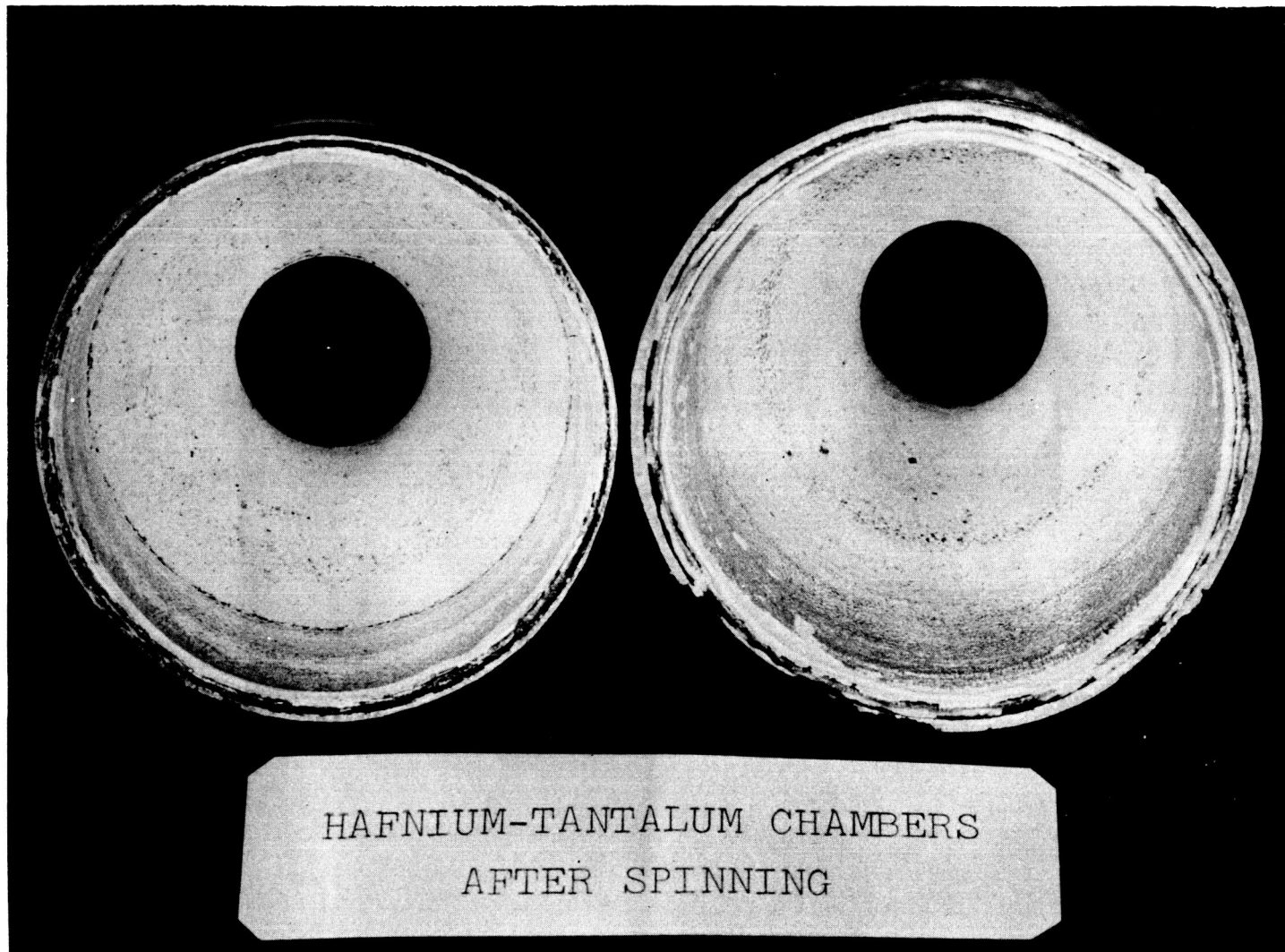


Figure V-149. Hafnium - Tantalum Chambers After Spinning

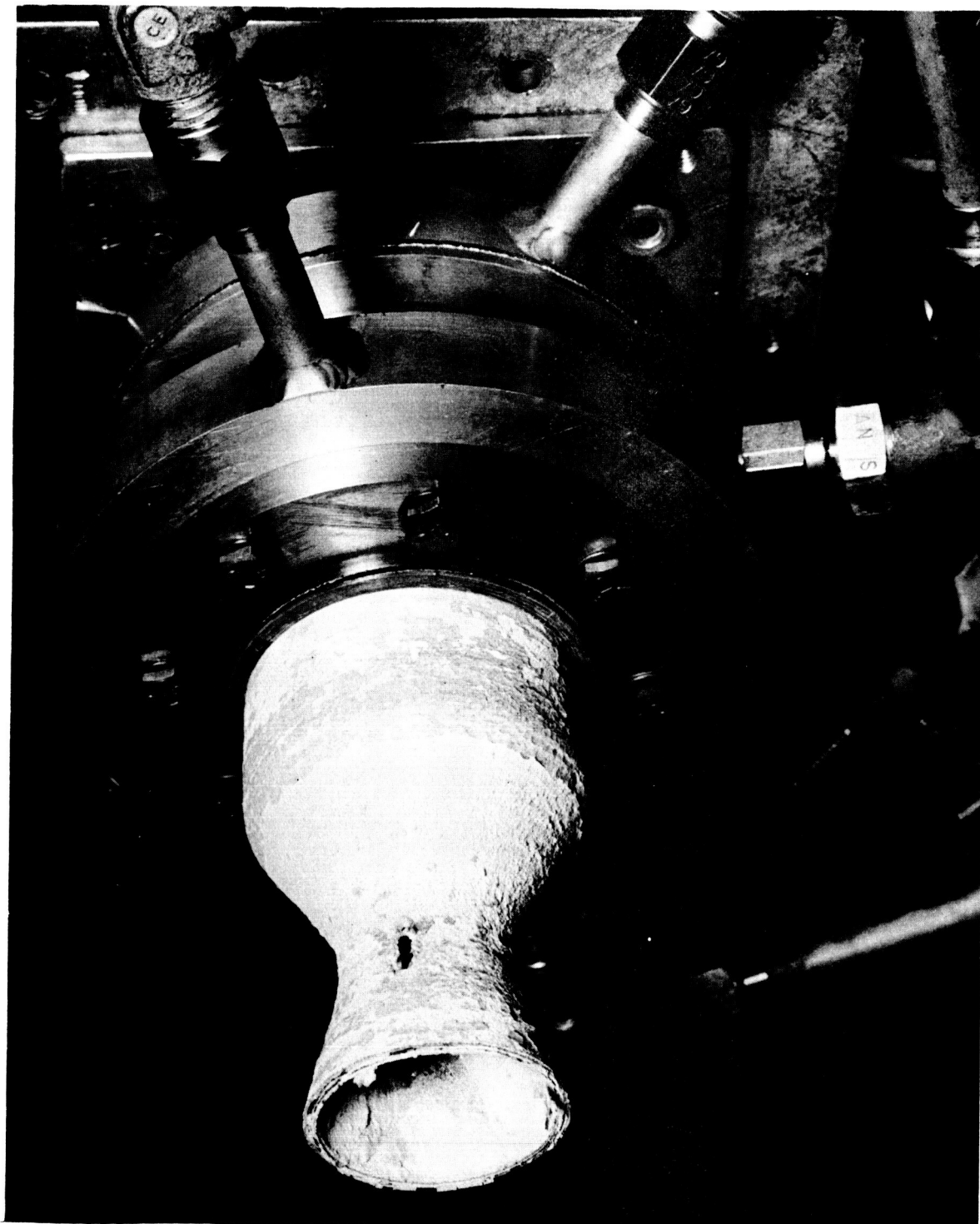


Figure V-150. Hafnium - Tantalum Clad Chambers After Fire Test

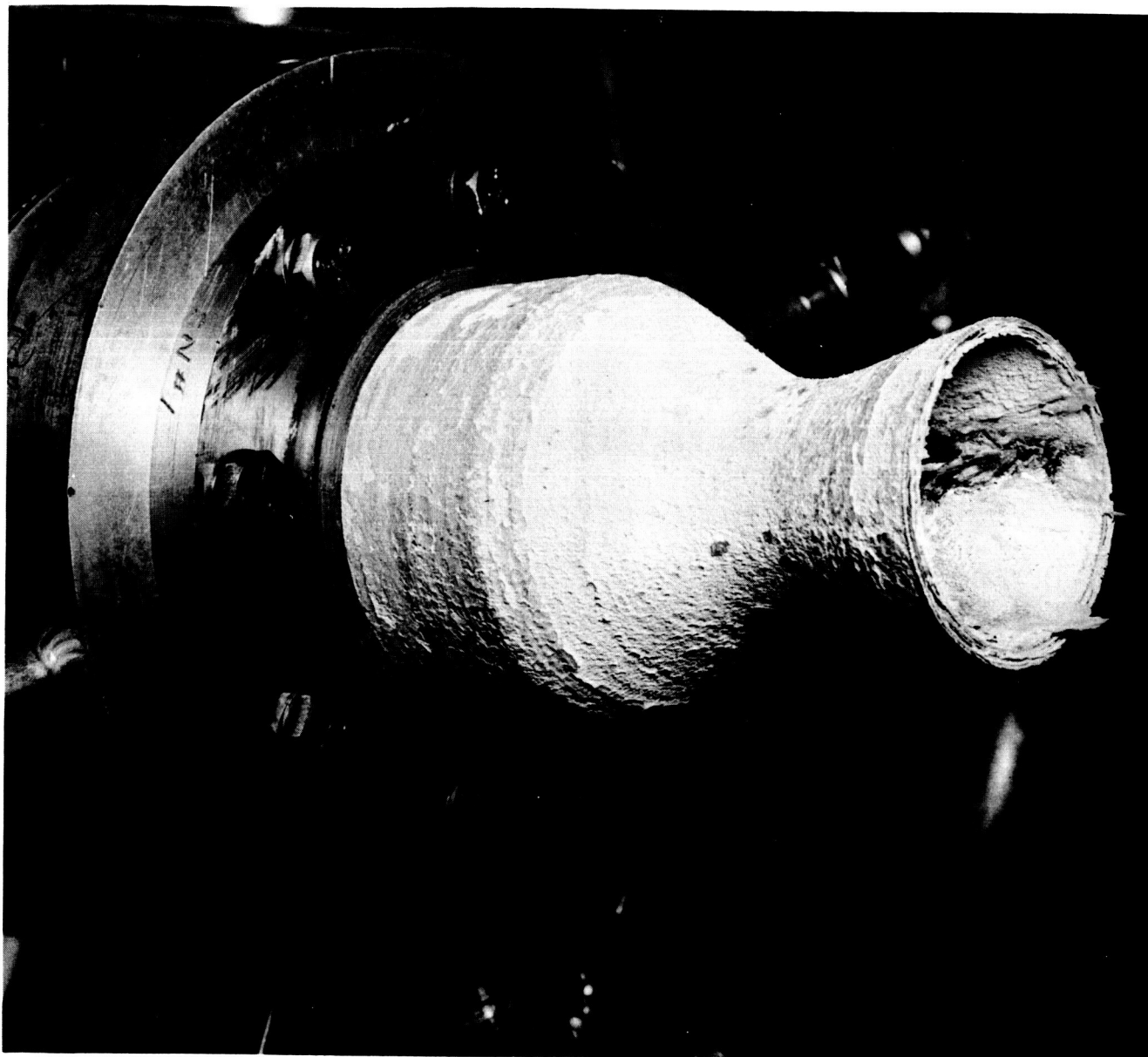


Figure V-151. Hafnium - Tantalum Clad Chambers After Fire Test

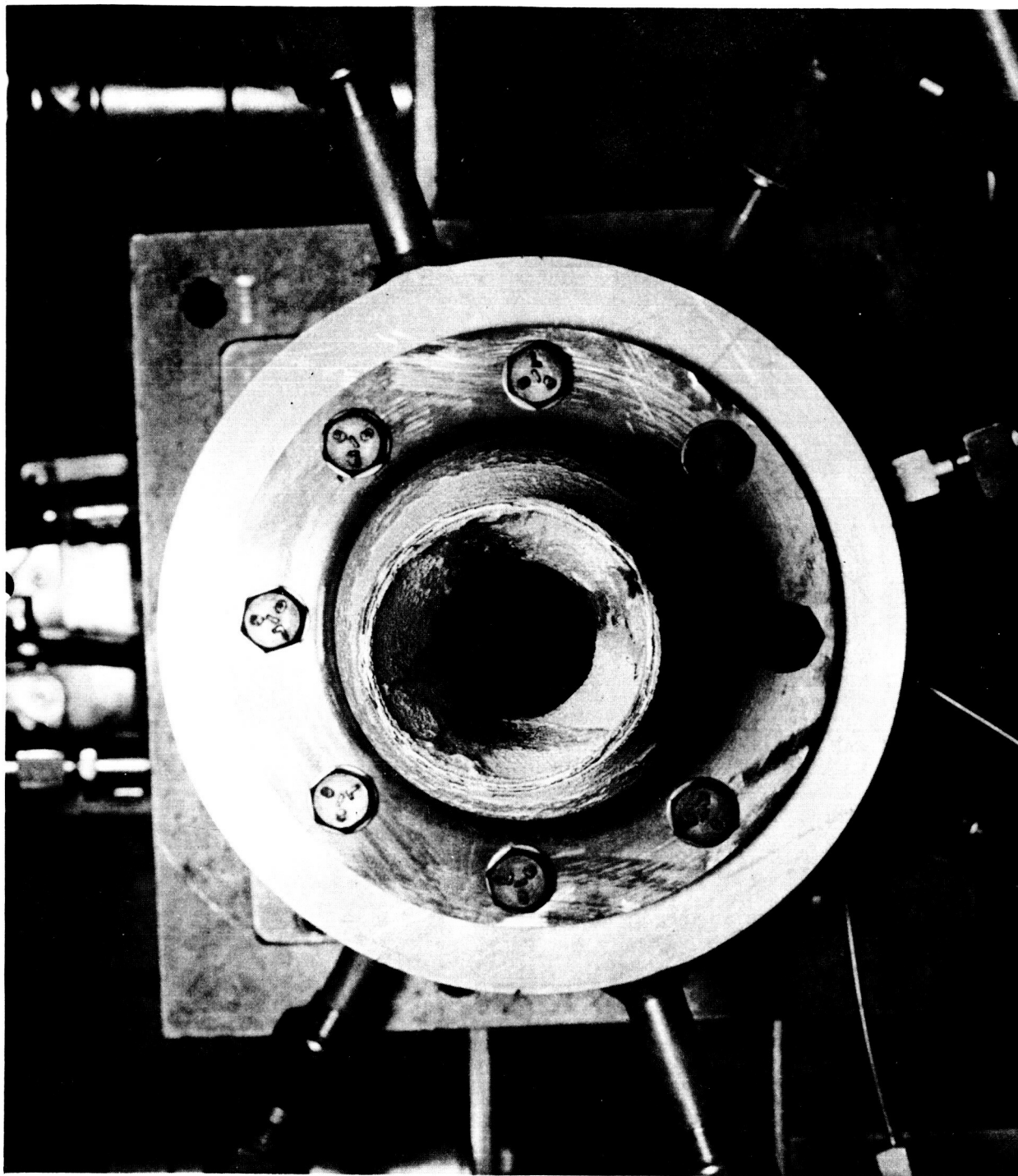


Figure V-152. Hafnium - Tantalum Clad Chambers After Fire Test

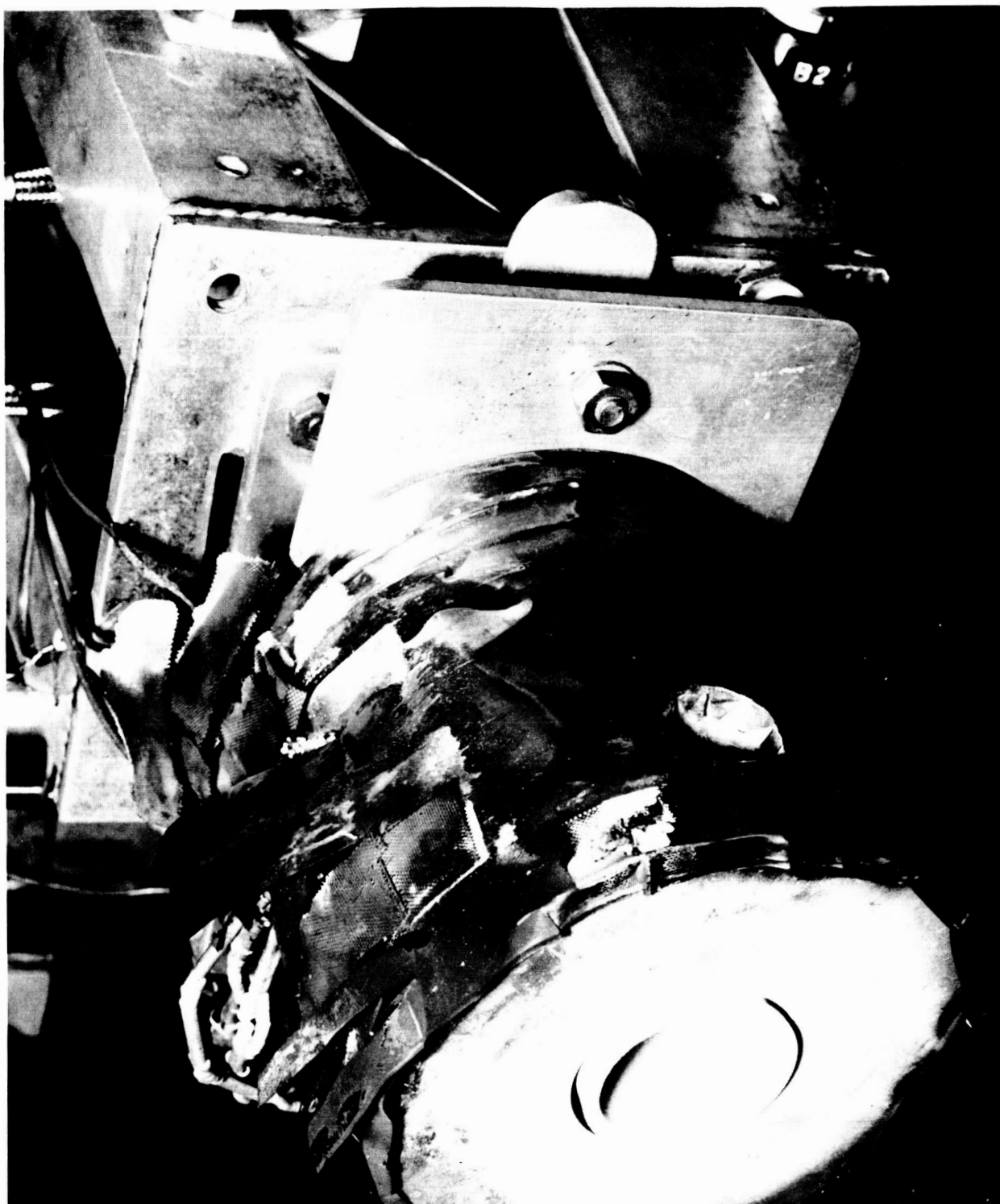


Figure V-153. Hafnium - Pantalum (Slurry Coated) Chamber After Biaxial Test

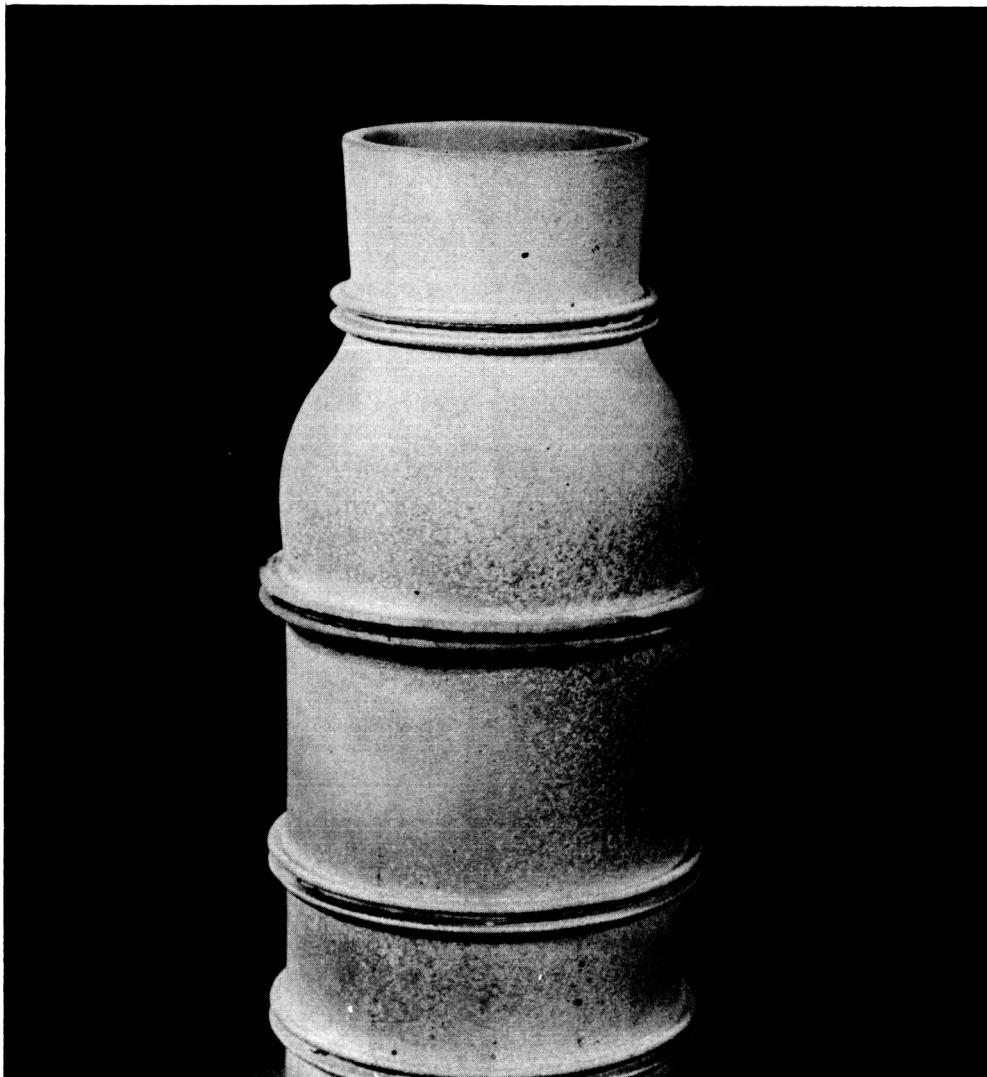


Figure V-154. Hafnium - Tantalum Coated Chamber (Columbium) Sylcor Slurry Process
Accumulated Fire Test Time of 285 Sec



Figure V-155. Hafnium - Tantalum Coated Chamber (Columbium) Sylcor Slurry Process
Accumulated Fire Test Time of 285 Sec

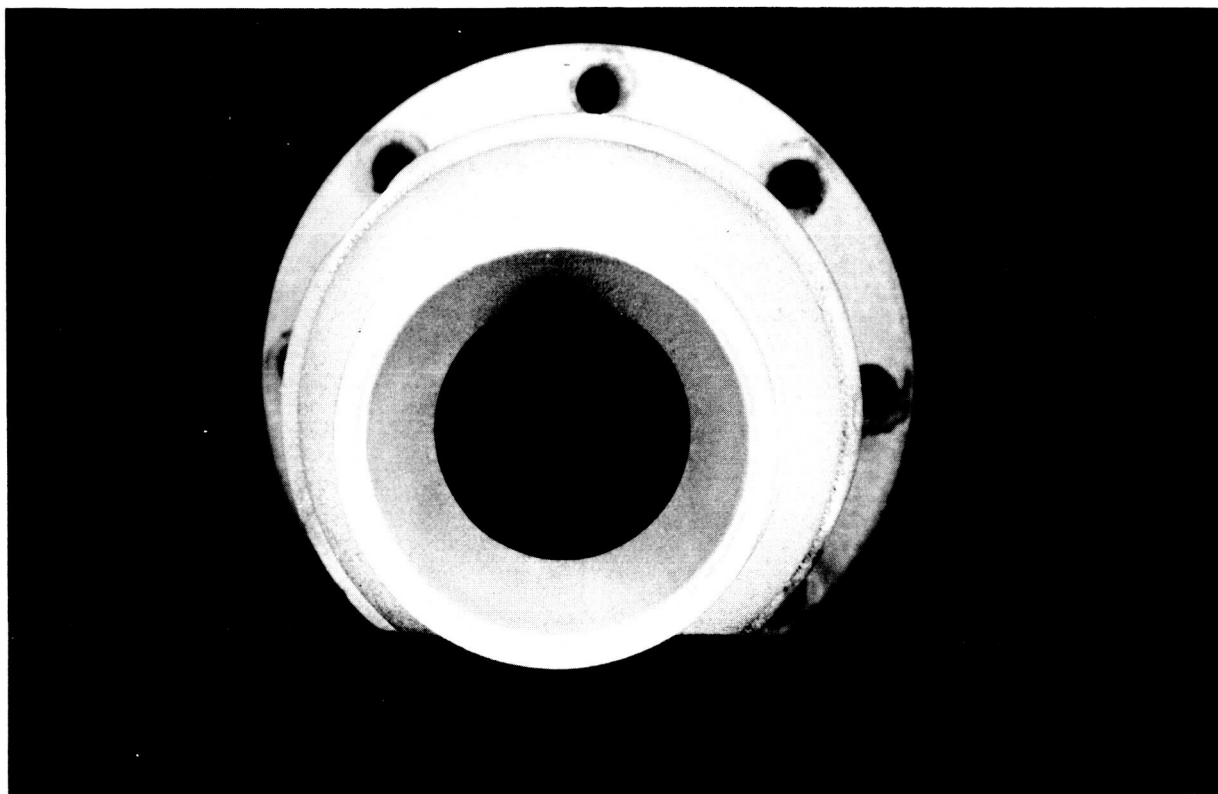


Figure V-156. Hafnium - Tantalum Coated Chamber (Columbium) Sylcor Slurry Process
Accumulated Fire Test Time of 285 Sec

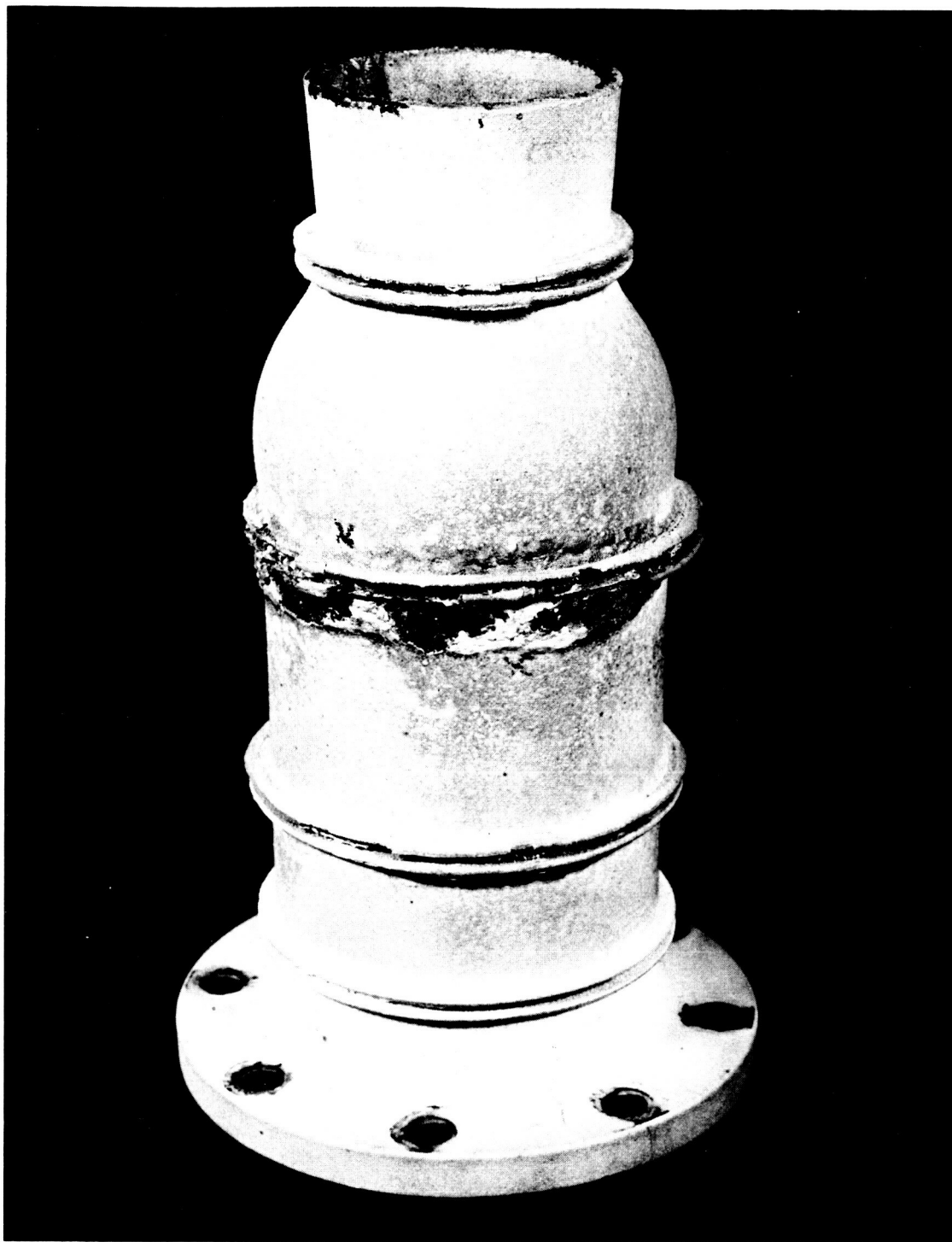


Figure V-157. Hafnium - Tantalum Coated Chamber (Columbium) Sylcor Slurry Process
Accumulated Fire Test Time of 424 Sec

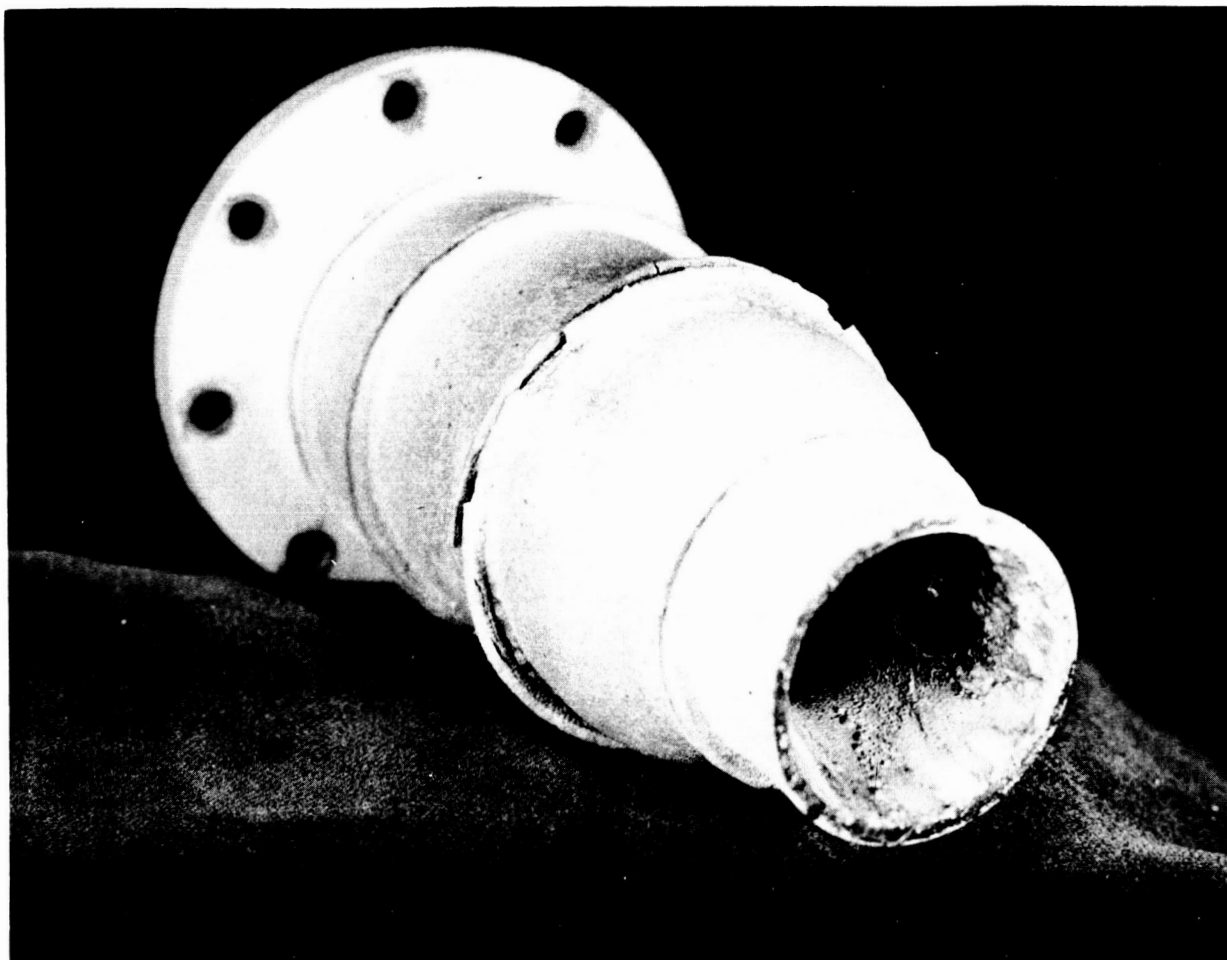


Figure V-158. Hafnium - Tantalum Coated Chamber (Columbium) Sylcor Slurry Process
Accumulated Fire Test Time of 424 Sec

F. INSULATION EVALUATION

High temperature insulation systems were investigated for applications to the prototype engine design. Tables V-20 and V-21 list the various insulation materials that were studied initially. Of these insulation materials three were selected based on their high temperature capability, thermal conductivity properties, density, predicted compatibility with the coating systems, and packaging characteristics. A thermal analysis discussed previously showed the feasibility of a composite insulation system. Alumina bubbles (aluminum oxide spheres) and graphite fibers were selected as the high temperature (3500°F) materials and Dyna-Quartz (modified Quartz Felt) as the lower temperature and lower thermal conductivity material.

1. Temperature Capability

Laboratory evaluation tests were conducted on graphite fibers (H.I. Thompson Company) and alumina bubbles (Carborundum) to determine the stability and integrity at anticipated temperature-pressure environments. Each material was evaluated at 10 μ , 100 μ and 3500 μ pressure at approximately 3500°F for one hour.

a. Results

The graphite fibers experienced no observable deterioration during the test exposures. After testing, the fibers were soft and flexible and showed no powdering. There was very little compacting of the fibers. There were no visual differences in the fibers exposed at the three different pressures. However, weight loss data indicates that some oxidation may have occurred, and that this oxidation increased with the increase in test pressure.

The hollow alumina bubbles experienced a small degree of melting at the interface with the hot heater rod. This indicates a melting point for the bubbles of approximately 3500°F. The bubbles immediately adjacent to the zone of melting did not melt. There were no observable differences in the bubbles exposed to different pressure levels. Some clumping of bubbles occurred near the hot zone; however, this was limited in scope and should present no serious problems during possible service use.

TABLE V-20
HIGH TEMPERATURE INSULATION MATERIALS

Material	Thermal Conductivity Data			Density (lb/ft ³)	Specific Heat BTU/lb-°F	Max. Use Temp (°F)	General
	Hot Surface Temp (°F)	Mean Temp (°F)	K Value (BTU·in./ft ² ·hr·°F)				
Grafoil Laminate (High Temperature Materials, Inc.)		1000 2000	2.4 - 10.8 2.4 - 4.8	10-100		≈ 5000	Cheaper than pyrolytic graphite.
Pyrolytic Graphite (Pyrogenics, Inc.)		80 2000	14.4 7.2	138		≈ 5000	
Pyrolytic Graphite (Various)		1000 2000 3000 3600	7.0 - 8.5 5.2 - 6.5 5.3 - 7.5 6.5 - 9.0			≈ 5000	Data by Southern Research Institute
Pyrolytic Graphite (Various)		1000 2000 3000 3600	10.3 7.5 4.8 3.1	138	1000F=0.038 2000F=0.059	≈ 5000	Data by R.G. Bourdeau C _p data seems low (Test method not given)
Graphite Fibers (Various)	1800 3600	1000 1800	0.4 0.77	9.5		≈ 5000	10 μ fibers;
Thoria Powder (Various)		2000 3140	0.42 0.93	0.26 g/cm ³			
Zirconia Fibers (H.I. Thompson Co.)		1000 1400	0.90 1.74	9		3000	Data at atmospheric pressure in air.
Alumina Foam (Various)	1900 3600	1000 1800	0.65 2.10	30		3600	
Graphite Fiber Felt (National Carbon Co.)			264		80F=0.16 2700=0.40	≈ 5000	
Alumina Bubble-Graphite Fiber System (Various)	1950 3500	1000 1770	0.32 1.4			3500	
Alumina Bubbles	3500	4.0		35		3500	

TABLE V-21
INTERMEDIATE TEMPERATURE INSULATION MATERIALS

Material	Thermal Conductivity Data			Density (lb/ft ³)	Specific Heat BTU/lb - °F	Max. Use Temp (°F)	General
	Hot Surface Temp (°F)	Mean Temp (°F)	K Value (BTU x in./ft ² hr x °F)				
Micro Quartz Type II (Johns Manville) (Dyna-Quartz)		1000 2000	0.70 1.56	9	400F=0.229 1800F=0.292	2750	Data determined at atmospheric pres- sure in air.
Micro Quartz Type II and Platinum Foils (Various)		1000 2000	0.20 0.41	6		2750	Data at 8 mm Hg.
High Temperature Super Insulation (Linde Co.)		1400	0.013	18		1900	Insulation package at high vacuum
Min-K 2000 (Johns Manville)		1000 1600	0.30 0.46	20	1000F=0.255 1800F=0.275	2000	Data determined at atmospheric in air.
Min-K 2000 (Johns Manville)		1000 1600	0.48 0.65	20		2000	Data determined at atmospheric pressure in helium.
Monsanto Micria ZR (Ziroconia Flakes and Shells) (Monsanto Chemical Co.)		2000	0.26	0.17 g/cm ³			Large shrinkage upon heating.
Silicon Carbide Foam		2480 800 1900	0.38 6 8	16-35		3000 in. oxid. atmos- 4000 in. inert atmos.	
Potassium Titanate Fibers (DuPont "Tipersul")		400 1000	0.28 0.36	28			
Potassium Titanate Block (DuPont)		500 1000	0.38 0.53	12		2200	
Flexible Min-K Tape		500 1000 600	0.29 0.35 0.22	28		1800	
		1000	0.30	Size: 2 1/2 x 1/4 in. Wt=0.12 lb/linear ft			
Fiber Frax GH Board (Carborundum Co.)		1600 1000 2000	0.44 0.80 1.40	30-40		2300	

Both insulation materials performed satisfactorily. An alumina bubble system offers an advantage in being oxidation resistant at high temperatures. However, the graphite fibers offer lower thermal conductivity and density and are easier to handle, and are serviceable at higher temperatures.

b. Test Procedure

The insulation materials were placed in a quartz tube, 3 inches long x 25/32 inch diameter. A solid tungsten or solid graphite heater rod, 1-1/8 inch long x 3/8 inch diameter, was centered in the tube with the insulation around it. A graphite rod was used for the graphite fiber test while tungsten was used for the alumina bubbles. A foamed zirconia block supported the heater rod and insulation at the bottom. The top of the rod was left exposed with the insulation leveled off at that point. The insulation, the heater rod, and the quartz tube and zirconia foam blocks were weighed before and after testing.

The heater rod was heated by a high frequency tube generator (Model T-25-3, Lepel High Frequency Laboratories). The quartz tube with the rod and insulation were installed in a water cooled vacuum chamber. The induction work coil was placed around the quartz tube. The pressure in the chamber was reduced and held at the required level.

The heater tube was then heated to the required temperature. The temperature was measured with a Leeds and Northrup optical brightness pyrometer (Model 8622). Emissivity corrections were made to insure an accurate test temperature. An exposure period of one hour was used for each test.

2. Compatibility Tests

Preliminary laboratory tests were performed with graphite fibers and alumina bubbles to determine their compatibility with hafnium-tantalum, tantalum-tungsten, tantalum, and columbium. The tests were conducted at 1 μ pressure and at 3200°F for 1/2 hour. A sample of each of the four refractory metals was installed in two ceramic crucibles and covered entirely with either graphite fibers or alumina bubbles. The results of the above evaluation are as follows:



	<u>Graphite Fiber</u>	<u>Alumina Bubbles</u>
Hafnium-tantalum	Fair	Fair
Tantalum-tungsten	Fair	Good
Tantalum	Poor	Poor
Columbium	Poor	Good
Poor	- oxidation or decomposition of the refractory metal noted.	
Fair	- slight discoloration of refractory metal.	

Further evaluation of the compatibility of carbon fibers, alumina bubbles, and Dyna-Quartz with the columbium alloys and Hf-Ta clad on a Ta-10%W substrate was continued. Although silicide coated SCb 291 columbium alloy was preferred, all silicide coated specimens were 752 columbium, due to availability. The chemical composition of the 508-C silicide coating is very similar, whether applied to 752 Cb or SCb 291.

These alloys were exposed to temperatures of 2600, 3000 and 3200°F with and without the three insulation materials. Bend tests were also performed.

The test conditions and observations of these compatibility tests are shown in Table V-22. The results of the bend tests are given in Table V-23.

Based on the metallurgical examination and bend tests of the exposed specimens, the following conclusions were reached:

- (a) The columbium alloy when coated with the 508-C silicide coating is more compatible with the insulation materials than when uncoated.
- (b) The Dyna-Quartz and alumina bubbles are more compatible with the coated columbium alloy than are the carbon fibers. Dyna-Quartz appears to be slightly more compatible than the alumina bubbles.
- (c) The compatibility of the Hf-Ta clad material with the insulation materials is somewhat questionable, possibly due to the effect of the uncoated edges.
- (d) In some cases, where the external surfaces appeared unattacked and shiny, the material had still become embrittled, as indicated by the bend tests.
- (e) The columbium alloy when uncoated became more embrittled than when coated.

TABLE V-22
LIST OF COMPATIBILITY TESTS - TEST CONDITIONS AND OBSERVATIONS

Test No.	Sample	Insulation	Test Temp °F	Test Atmosphere	Specimen Examination	
					Visual (1X)	Microscopic (100X)
1	SCb-291 Bare	None	3200	Vacuum 1 x 10 ⁻⁵ mm	Very bright and shiny, no scale or surface reactions	Very good - slight oxide penetration at surface
**2						
3	Hf-Ta (clad) on Ta-10W core	None	3200	Vacuum 1 x 10 ⁻⁵ mm	Very bright, shiny, no evidence of any reaction	None
4	Hf-Ta (Clad) on Ta-10W core	Al ₂ O ₃ Bubbles	3200	Vacuum 1 x 10 ⁻⁵ mm	Dull, whitish, surface evidence of reaction undercutting of clad layer, dimpled surface	None
5	SCb-291 Bare	Al ₂ O ₃ Bubbles	3200	Vacuum 1 x 10 ⁻⁵ mm	Bright, slight matte finish, no surface reaction	Very good - slightly more oxide penetration than test No. 1
6	*Coated Cb-752	None	3000	Argon	Dark-coating flaking off.	No oxidation of base metal, some penetration into coating
7	*Coated Cb-752	Al ₂ O ₃ Bubbles	3000	Argon	Whitish-blue appearance gross edge effect (melting) undercutting of coating, flaking.	Melting at the interface
8	*Coated Cb-752	Dyna-Quartz	2600	Argon	Slightly darker gray than as received, some small bubbles on surface	None

TABLE V-22 (CONT)

Test No.	Sample	Insulation	Test Temp °F	Test Atmosphere	Specimen Examination	
					Visual (1X)	Microscopic (100X)
9	SCb-291	Dyna-Quartz	2600	Argon	Gray with heavier gray lines. No reactions at the surface, only gray discoloration	None
10	*Coated Cb-752	Carbon fibers	3200	Argon	Flaking of coating - signs of melting under coating reaction with coating and fiber-coating. Very brittle.	Sign of melting penetration into the base metal; grain boundary reaction with coating and fibers.
11	SCb-291 Bare	Carbon fibers	3200	Argon	Gray with imprints of fiber embedded. No gross reaction.	Penetration at surface - some intergranular penetration - more severe than Al ₂ O ₃ bubbles.
12	Hf-Ta (Clad) on Ta-10W	Carbon fibers	3200	Vacuum 1 x 10 ⁻⁵ mm	Gross attack at surface, complete destruction of core material.	None
13	Cb-752	None	3000	Argon	Bright, shiny, no reaction	None
14	*Coated Cb-752	One side Al ₂ O ₃ other side carbon fibers	3000	Argon	Surface reaction on both sides melting and undercutting of coating - both surfaces attacked severely.	None
15	SCb-291	Al ₂ O ₃	3200	8 mm Air	Blackened and dimpled surface; minor attack.	None

TABLE V-22 (CONT)

Test No.	Sample	Insulation	Test Temp °F	Test Atmosphere	Specimen Examination	
					Visual (1X)	Microscopic (100X)
16	SCb-291	Al ₂ O ₃ Bubbles	3000	8 mm Air 1 hour	Blackened surface	Gross attack at edges complete penetration of sample grain boundary attack. Very brittle. Shattered when cutting.
17	*Coated Cb-752	Al ₂ O ₃ Bubbles	3000	8 mm Air 1 hour	Bright - appeared unchanged	Some grain growth coating afforded protection
18	SCb-291 Bare	None	3000	8 mm Air	Slightly gray - no pitting or defect apparent	Oxide penetration through complete sample. Very brittle, broke when cutting.
19	*Coated Cb-752	Al ₂ O ₃ Bubble	3000	8 mm Air	Grey in appearance No visual attack observed	Oxide penetration into sample. Grain and grain boundary affected.
20	*Coated Cb-752	Dyna- Quartz	3000	8 mm Air	Metallic gray coating appearance - unchanged	Grain growth - no attack (Sample appeared protected by coating).
21	*Coated Cb-752	Dyna- Quartz	3000	8 mm Air	Same as above	

* All samples were coated with GT&E modified silicide coating.

** Test No. 2 - Malfunction in heating. Data not recorded.

TABLE V-23
BEND TEST RESULTS

	<u>Sample</u>	<u>Results</u>
1	SCb-291 As received	Bent to form a "U"
2	SCb-291 Heated 1/2 hour in vacuum - no insulation	Brittle - broke without noticeable deflection
3	SCb-291 Heated 1/2 hour with Al ₂ O ₃ bubbles in vacuum	Brittle - broke without noticeable deflection
4	Coated Cb-752 GT & E modified silicide coating (as received)	Bent to 90° without fracture
5	Coated Cb-752 (GT & E modified silicide coating). Heated 1/2 hour in Argon with no insulation	Crack after slight deflection
6	SCb-291 - 1 hour in 8 mm insulated with Al ₂ O ₃ bubbles	Very brittle - shattered when cutting
7	Coated Cb-752 (GT & E modified silicide coating) 1 hour in 8 mm air insulated with Al ₂ O ₃ bubbles	Crack after slight deflection
8	Cb-752 alloy as received	Bent to "U" shape
9	Cb-752 Coated with GT & E modified silicide, insulated with Dyna-Quartz, in 8 mm air.	Bent to better than 90° without cracking

**G. PROPELLANT VALVE DESIGN****1. Vendor Survey**

Design requirements were prepared as follows and issued to Moog, Parker, National Water Lift, Sterer, and Hydraulic Research as proposal requests:

<u>Bipropellant Valve Design Attribute</u>	<u>Bell Aerosystems Requirements</u>
1. Operating Pressure	300 psig
Proof Pressure	450 psig
Burst Pressure	600 psig
2. Operating Fluids	
Fuel	MMH
Oxidizer	N_2O_4
3. Energized Time	3600 seconds of continuous operation at 30 VDC
4. Response Time	
Open	0.005 second maximum at 18 VDC
Closed	0.005 second maximum at 18 VDC
5. Cycle Life	50,000 cycles minimum over period of one year
6. Rated Flows	
Fuel	0.14 pound/second
Oxidizer	0.20 pound/second
7. Pressure Drop	50 psi each side (assumed to be maximum)
8. Operating Voltage	18 - 30 VDC
9. Power Requirements	Keep to minimum



<u>Bipropellant Valve Design Attribute</u>	<u>Bell Aerosystems Requirements</u>
10. Downstream Volume	Low as possible
11. Weight	Low as possible
12. Internal Leakage	5 cubic centimeters/hour N ₂ gas mixture each side over entire cycle life
13. a. Valve Position	Normally closed
b. Simultaneous Operation	Within 0.001 second
14. Operating Temperature Range	-40°F to +160°F
15. Mounting Interface	
16. Redundant Features	
Coil	Required
Seats	Desirable

Proposals were received from Moog, Inc., Parker Aircraft Co. and National Water Lift Co. In addition, in conjunction with the work being done on another engine proposal for NASA-MSFC, two other proposals were received, one from National Water Lift and one from Vickers Incorporated. Drawings of all five valves are shown in Figures V-159 through V-163. Cutaway sketches of the Moog and Parker valves are given in Figures V-164 and V-165, respectively. It should be noted that the sketch of the Moog valve does not reflect the most current design.

2. Propellant Valve Rating

A complete rating of all five valve designs has been made. The rating system has a maximum score of 485 points and is shown in Figures V-166 and V-167, which also include the scoring breakdown of each valve design.

Using the rating system, the following scores were established:

Moog, Inc.	436
Parker Aircraft	414
Vickers	360
National Water Lift (Biprop.)	351
National Water Lift (Dual Sol.)	340

It is obvious that, the Moog and Parker valves are very comparable.

3. Propellant Valve Selection

The Moog bipropellant valve was selected for use on the prototype engines, based on cost considerations and results of tests by NASA-Houston on similar propellant valves.

The design requirements established for 100 pound thrust Moog bipropellant valve are as follows:

- (1) Nominal Working Pressure: 350 psig
Rated Operating Pressure: 500 psig
Proof Pressure: 750 psig
Burst Pressure: 1,000 psig
- (2) Operating Media:
Fuel: Monomethylhydrazine (S.G. = 0.87 at +70°F)
Oxidizer: Nitrogen Tetroxide (S.G. = 1.445 at +70°F)
- (3) Flow Rates:
Fuel: 0.14 lb/sec MMH at $\Delta P = 45 \pm 4$ psi
Oxidizer: 0.20 lb/sec N_2O_4 at $\Delta P = 45 \pm 4$ psi
- (4) Voltage:
18.0 to 30.5 VDC
- (5) Current: The maximum allowable current drain shall be 2.5 amperes.
- (6) Excitation Circuits: The valve shall not contain or require the use of electronic driving or switching circuits to meet the performance requirements herein.

- (7) Insulation Resistance: 100 megohms minimum at 500 ± 50 VDC.
- (8) Response: Maximum of 0.005 second to open or close at ambient atmospheric conditions with 350 psig inlet pressure and 28 VDC applied voltage. The valve shall operate with 500 psig inlet pressure and 18 to 30.5 VDC applied voltage.
- (9) Simultaneity: The fuel and oxidizer sides of the valve shall actuate simultaneously or within ± 0.001 second of each other.
- (10) Maximum "On" Time: 3600 seconds without any damage to the valve while flowing propellants per Item 3.
- (11) Leakage:
 - External: Zero when checked with nitrogen gas to five minutes (minimum) at the proof pressure (750 psig).
 - Internal: The leakage rate from either propellant section shall not exceed 5.0 scc/hr of nitrogen gas when checked for a minimum duration of 12 minutes with inlet pressures from 0 to 350 psig. The leakage rate with inlet pressures from 350 to 500 psig shall not exceed 8.0 scc/hr of nitrogen gas.
- (12) Cycle Life: 50,000 cycles over a one (1) year period.
- (13) Ambient Operating Temperature Range: -40°F to $+160^{\circ}\text{F}$
- (14) Soakback Temperature: The estimated soakback temperature at the valve outlet, due to engine firing, is $+260^{\circ}\text{F}$ (after shutdown).
- (15) Filtration: Each propellant section shall contain an integral stainless steel filter located between the inlet port and the shutoff point. The filters shall have ratings no higher than 20 microns nominal and 35 microns absolute. The total filter area shall be greater than 1.0 square inch.
- (16) Vibration:
 - Sinusoidal:
 - 5 - 47 cps at 0.052 in. double amplitude
 - 47 - 200 cps at 6.0 g peak
 - 200 - 260 cps at 0.0029 in. double amplitude
 - 260 - 2,000 cps at 10.0 g peak
 - The frequency range shall be scanned logarithmically at a rate of 1.0 octave per minute from 5 cps to 2,000 cps and back to 5 cps in each of three mutually perpendicular axes.

Random:

20 - 85 cps at $0.065 \text{ g}^2/\text{cps}$
85 - 280 cps at 6.3 db/octave
280 - 1,000 cps at $0.8 \text{ g}^2/\text{cps}$
1,000 - 2,000 cps at -12.0 db/octave
2,000 cps at $0.05 \text{ g}^2/\text{cps}$

Random excitation shall be for a duration of 12 minutes in each of three mutually perpendicular axes.

- (17) Shock: 20 g peak of half sine wave form for a 10 millisecond duration in each of three mutually perpendicular axes. In addition, the valve shall remain intact upon receiving a shock of 78 g applied in terminal sawtooth wave for an 11 ± 1 millisecond duration.
- (18) Acceleration: The valve shall operate during and after a sustained acceleration environment of 20 g for a 120 second duration applied in any of three mutually perpendicular axes.
- (19) Construction: The valve shall utilize welded joints as much as practical.
- (20) Porting: Each inlet port shall be a 1/4-inch tube size port similar to MS-24385, and shall be capable of accepting a Lee Jet orifice insert. The outlet porting shall consist of a flat faced, bolt-on manifold type connection.
- (21) Residual Volume: The residual propellant volume downstream of the valve seats shall be kept to a minimum consistent with the requirements herein.
- (22) Materials: Materials of metal parts which are in contact with the propellants shall be 304, 304L, 316, 17-4, 17-7 or stellite 25 corrosion resistant steel.
- (23) Fastening: All screw fasteners shall be locked by safety wire or other mutually approved methods.
- (24) Identification: Identification of the fuel and oxidizer sides of the valve shall be clearly and permanently marked on the valve.

4. Evaluation of a Welded Propellant Valve - Injector Interface

A review was made of welding the Moog bipropellant valve to the injector assembly. The Hamilton Standard Co., who design and manufacture electron beam welding equipment, and the Electron Standard Co., who do a considerable amount of electron beam welding, were contacted concerning several "novel" welding approaches. As a result, a highly simplified valve/injector joint has been generated. The joint, shown in Figure V-168, is produced with

only one weld pass or setup. In this concept, the electron beam, directed perpendicular to the two barrel sections in View A-A, is set up to pass through the barrels. This produces a weld joint completely surrounding each flow passage. Allowance for weld splatter in the flow passage is provided by locally enlarging the passage.

Figure V-168 also shows a method of complete welding of the Moog bipropellant valve. This is a highly practical method since it provides the capability of holding and removing the valve seat before welding. This is provided by an assembly plate which is not a part of the final valve assembly. When the seat fabrication and assembly is satisfactory (primary leakage considerations), the seat retainers are tack welded to the valve body through holes in the assembly plate. The plate is then removed and the complete weld is made.

5. Propellant Valves - 5 to 1000-Pound Thrust Range

The performance characteristics of bipropellant valves over a 5 to 1000-pound thrust range were studied and a list of some pertinent parameters is shown in Figure V-169.

The last item in Figure V-169 lists a throttle valve made by the Fox Valve Development Co. A detailed description of this valve is given in the subsequent section.

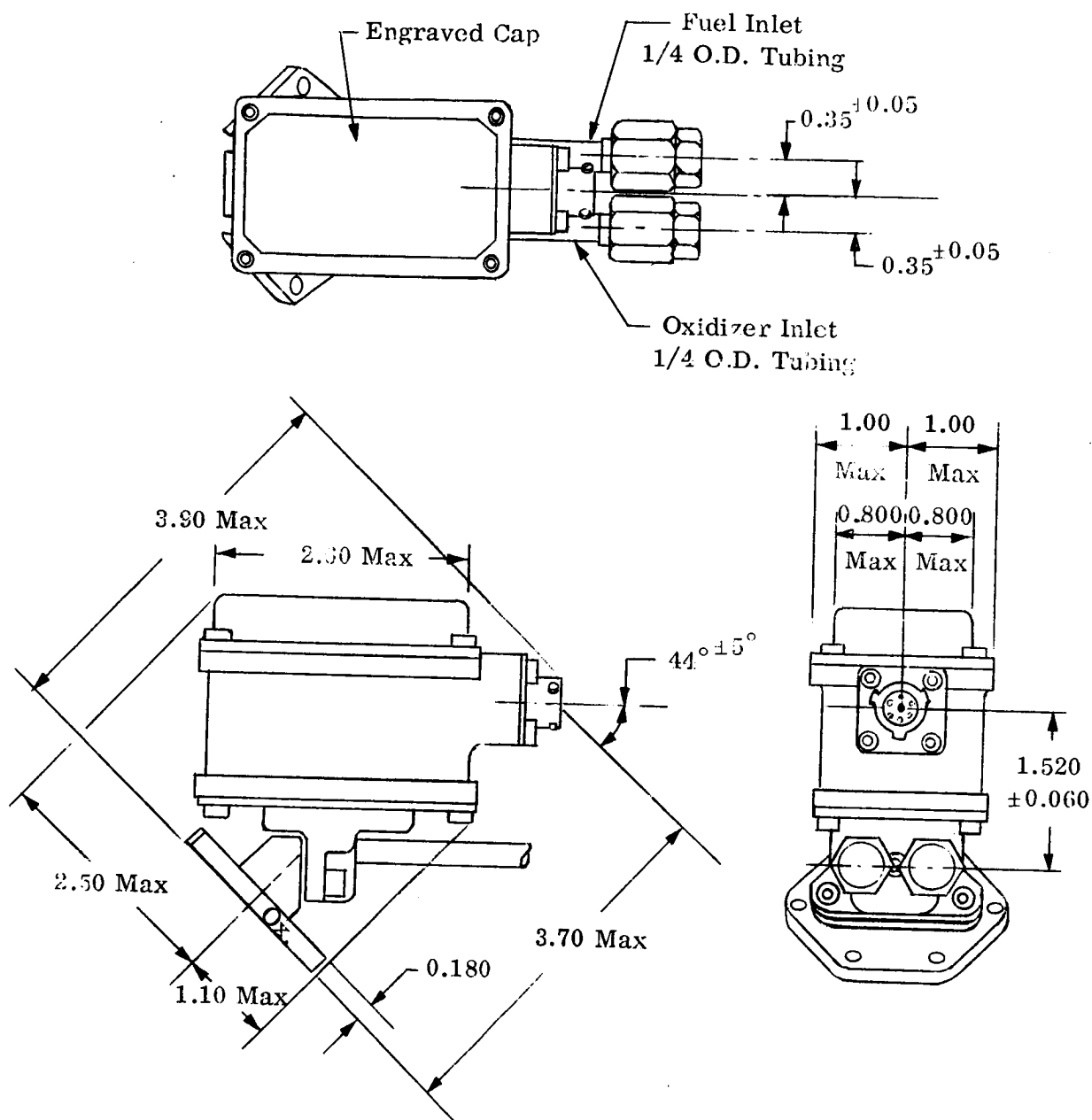


Figure V-159. Moog Bipropellant Valve

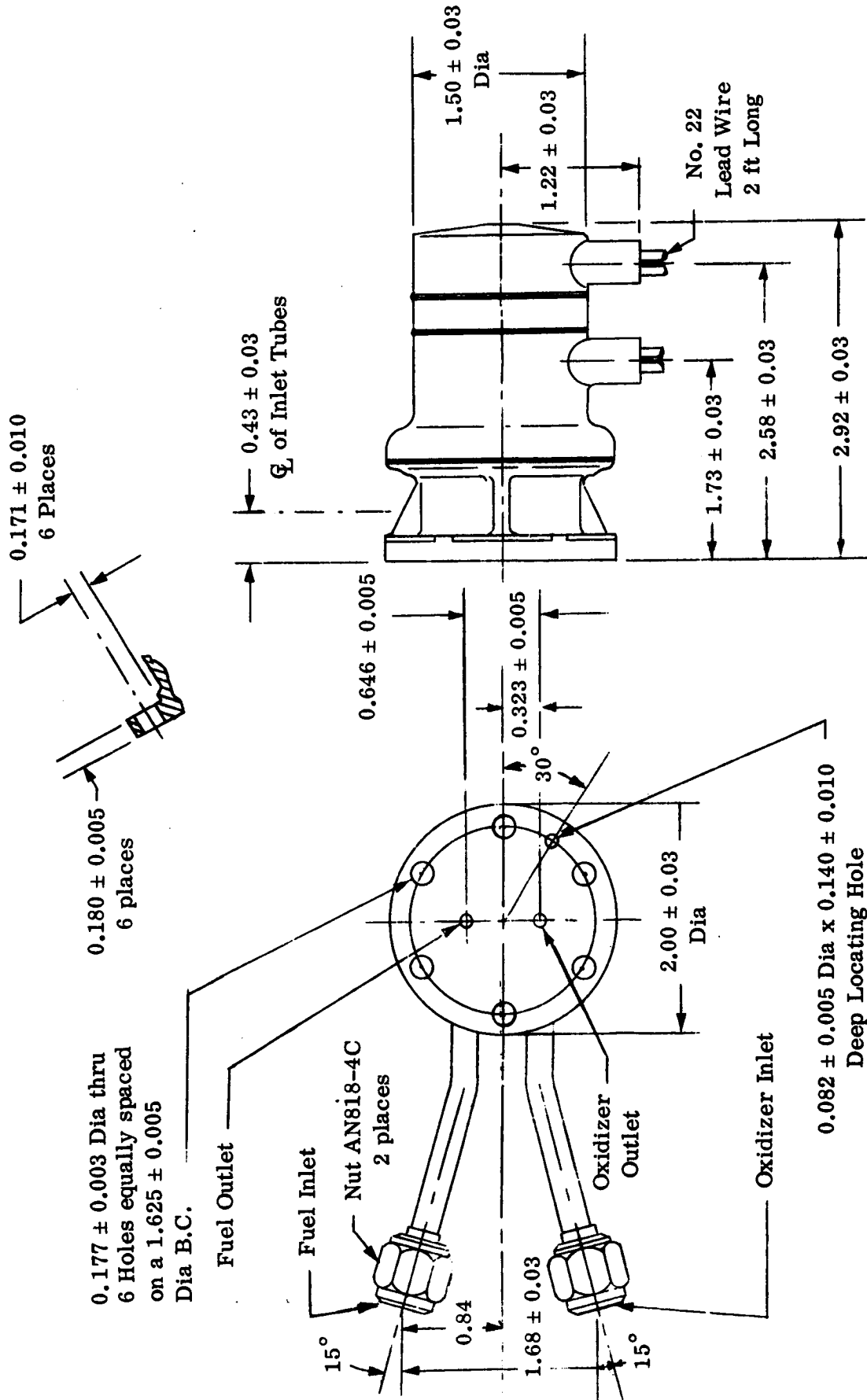


Figure V-160. Parker Pulse Operated Bipropellant Valve Mechanically Linked With Manual Coil

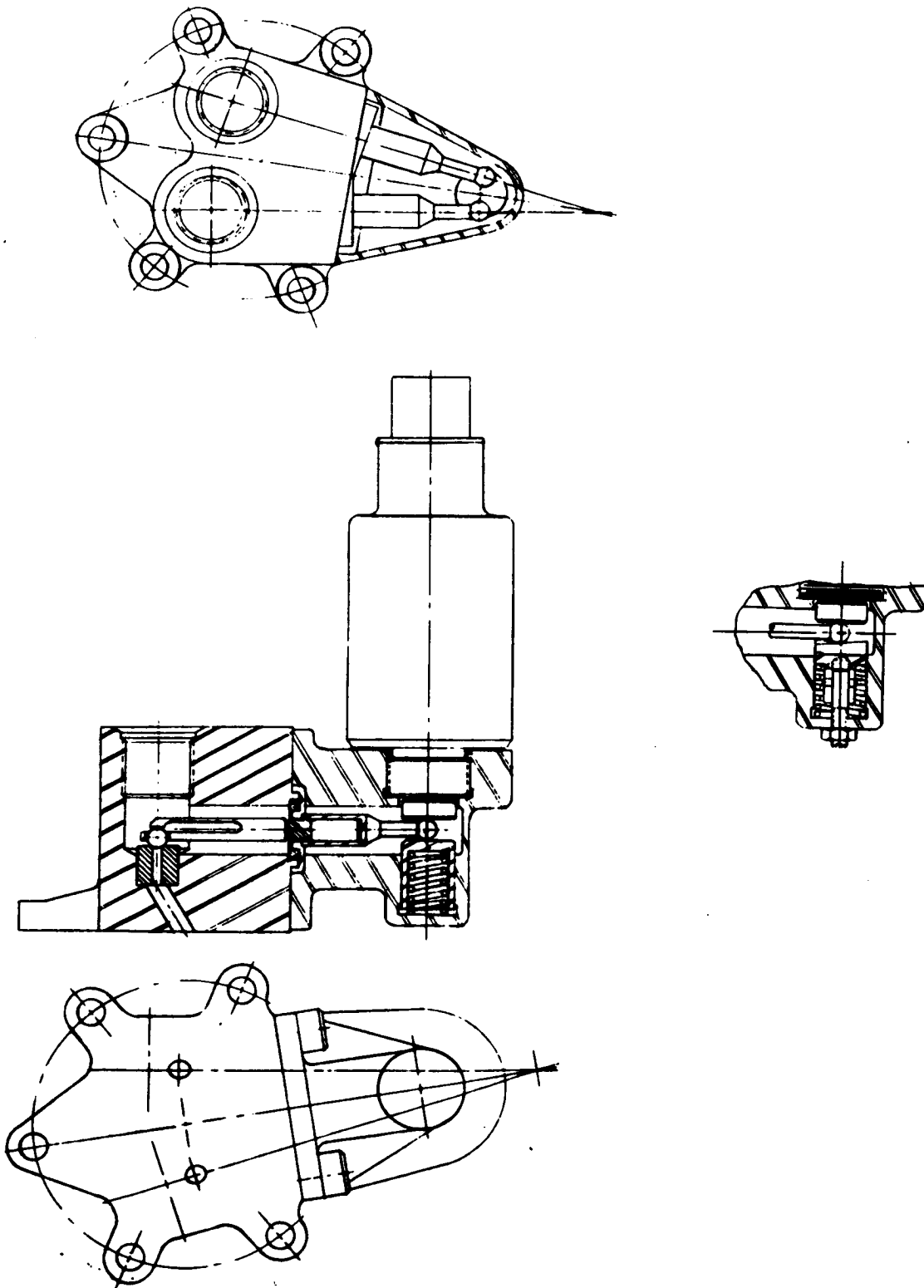
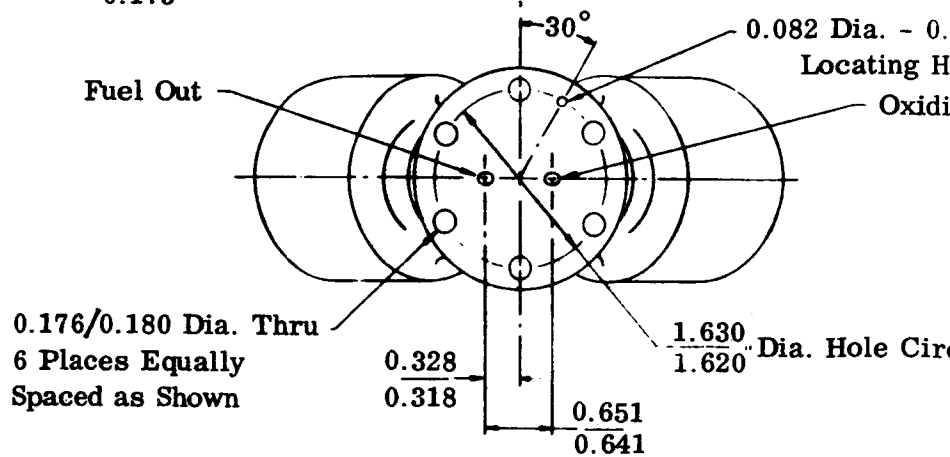
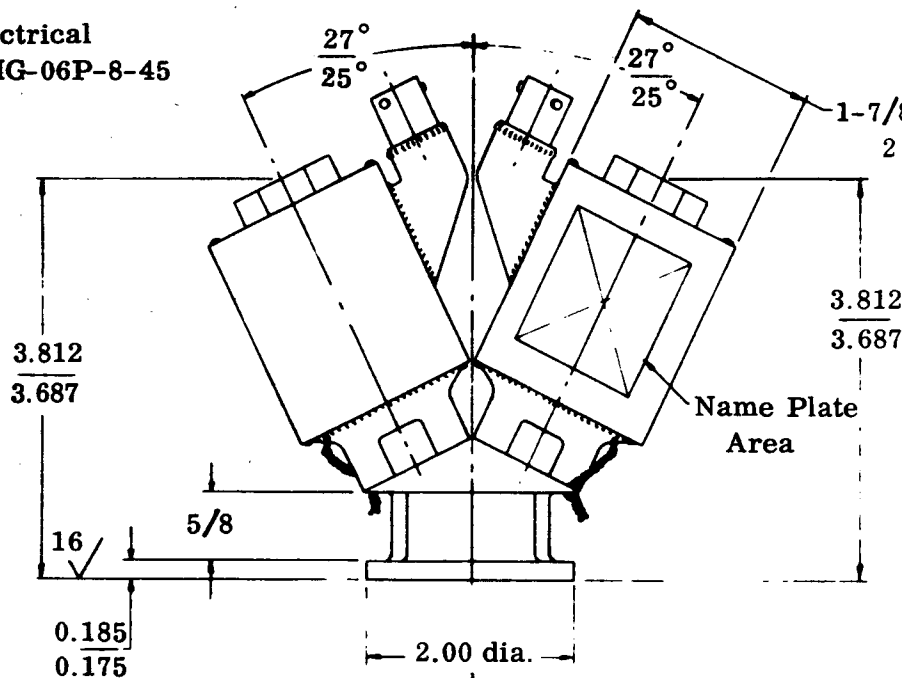
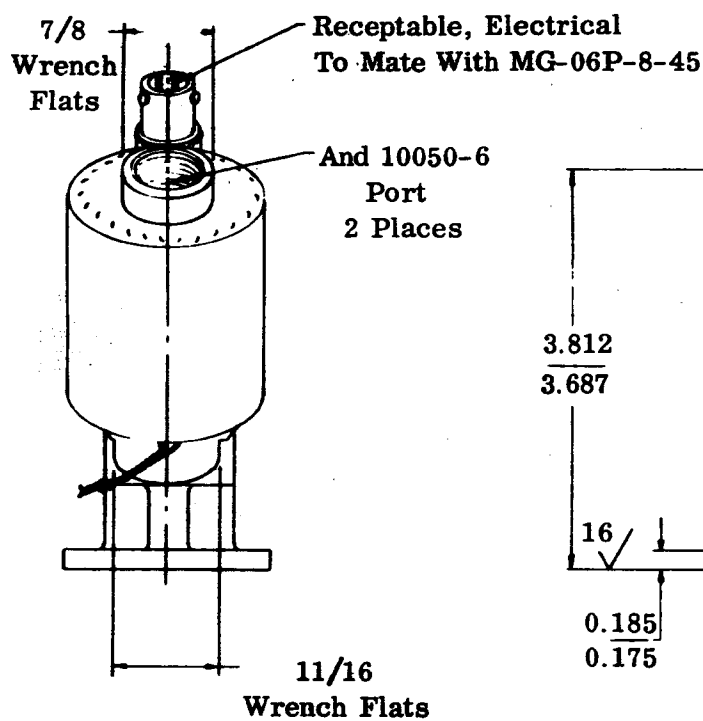


Figure V-161. National Water Lift Bipropellant Valve



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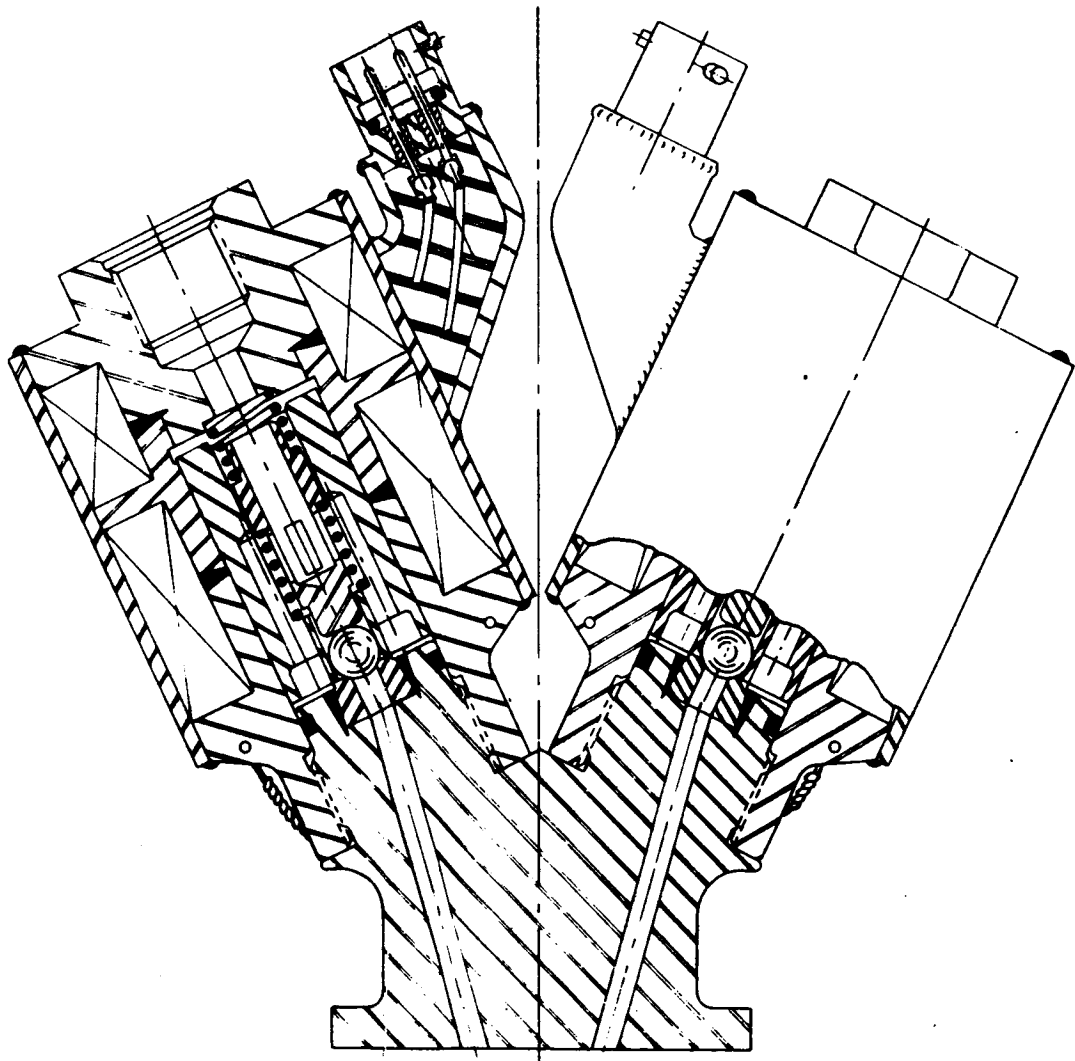
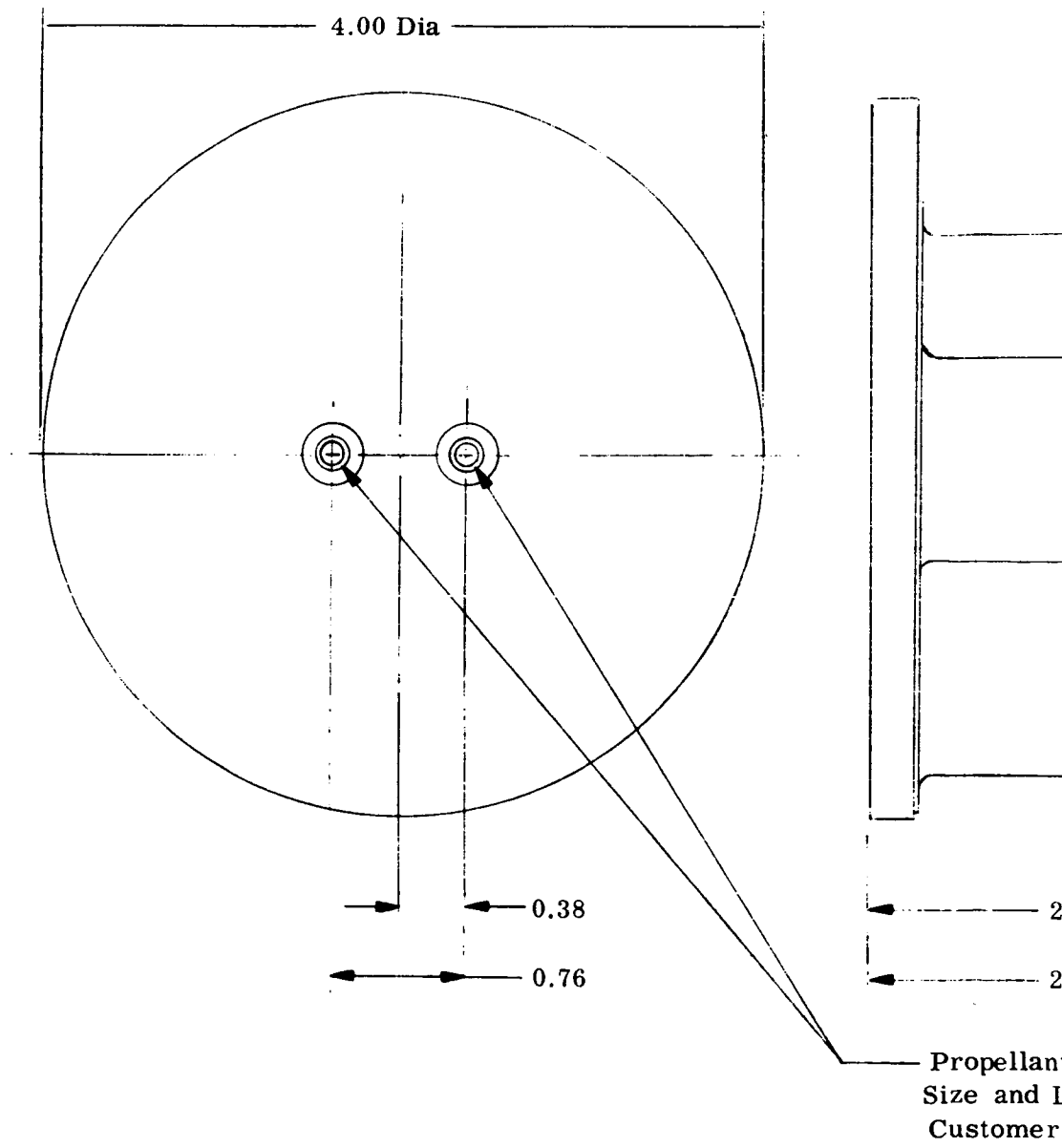


Figure V-162. National Water Lift Valve - Dual Solenoid

b



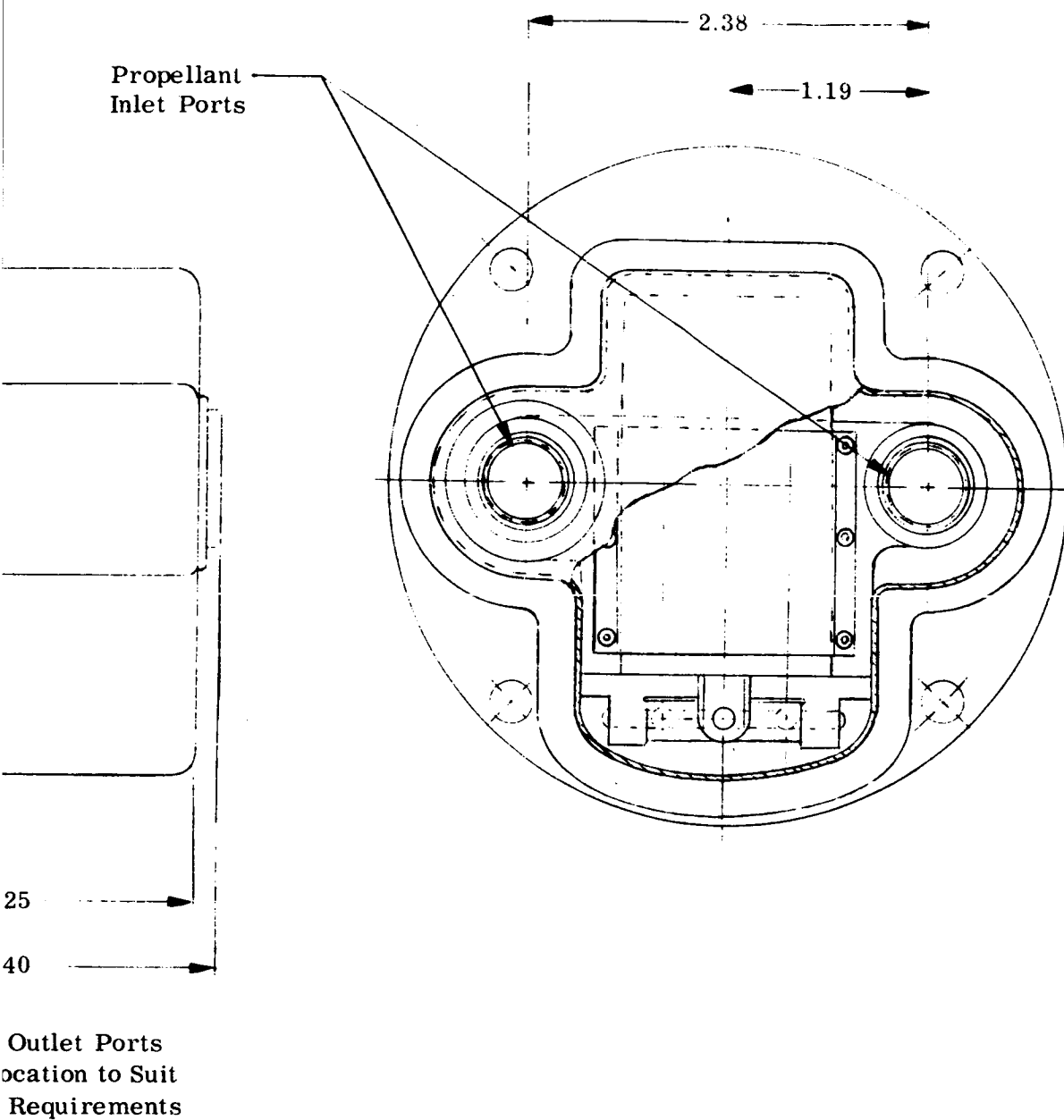
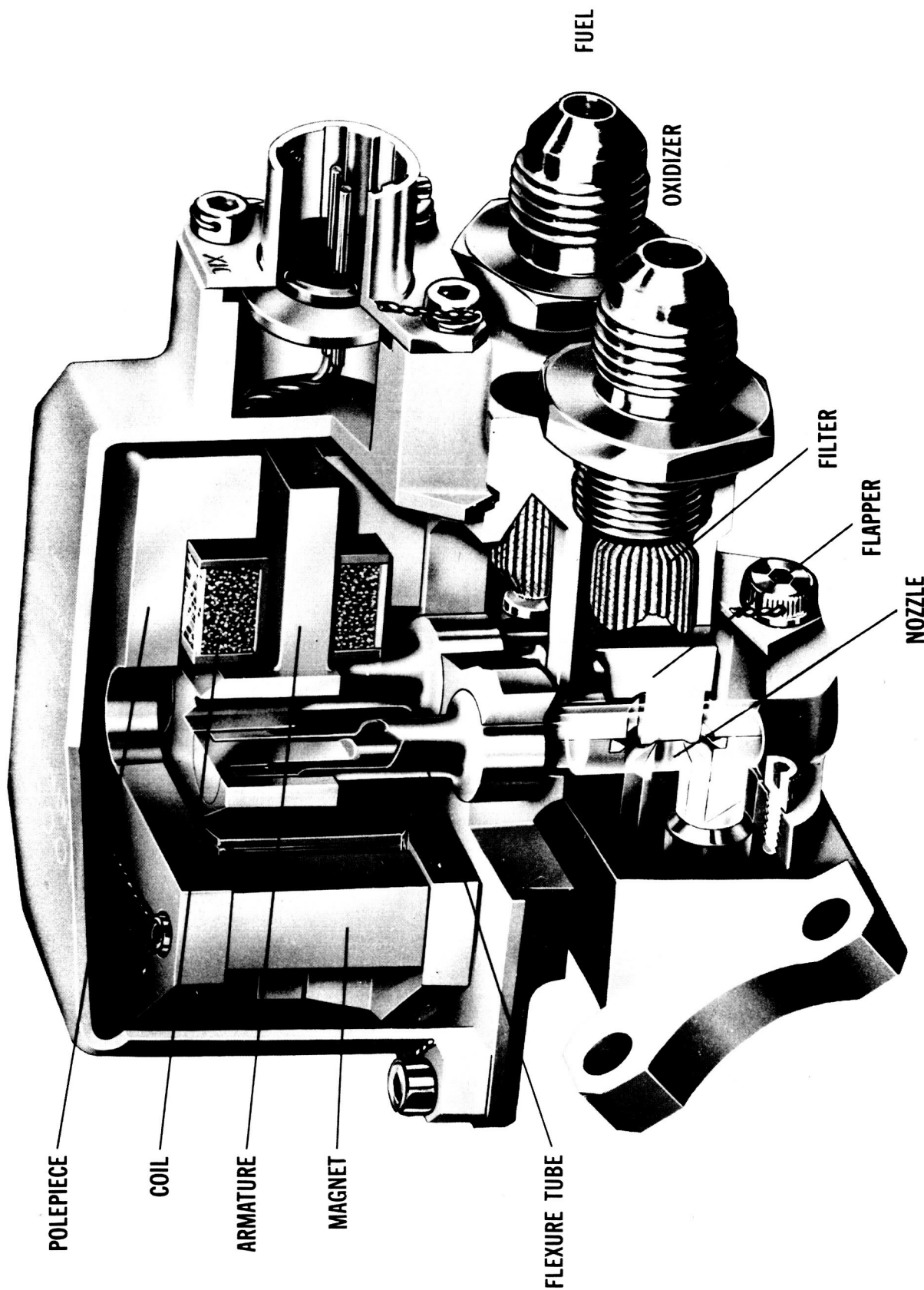


Figure V-163. Vickers Bipropellant Valve



R-125

Figure V-164. Moog Bipropellant Valve

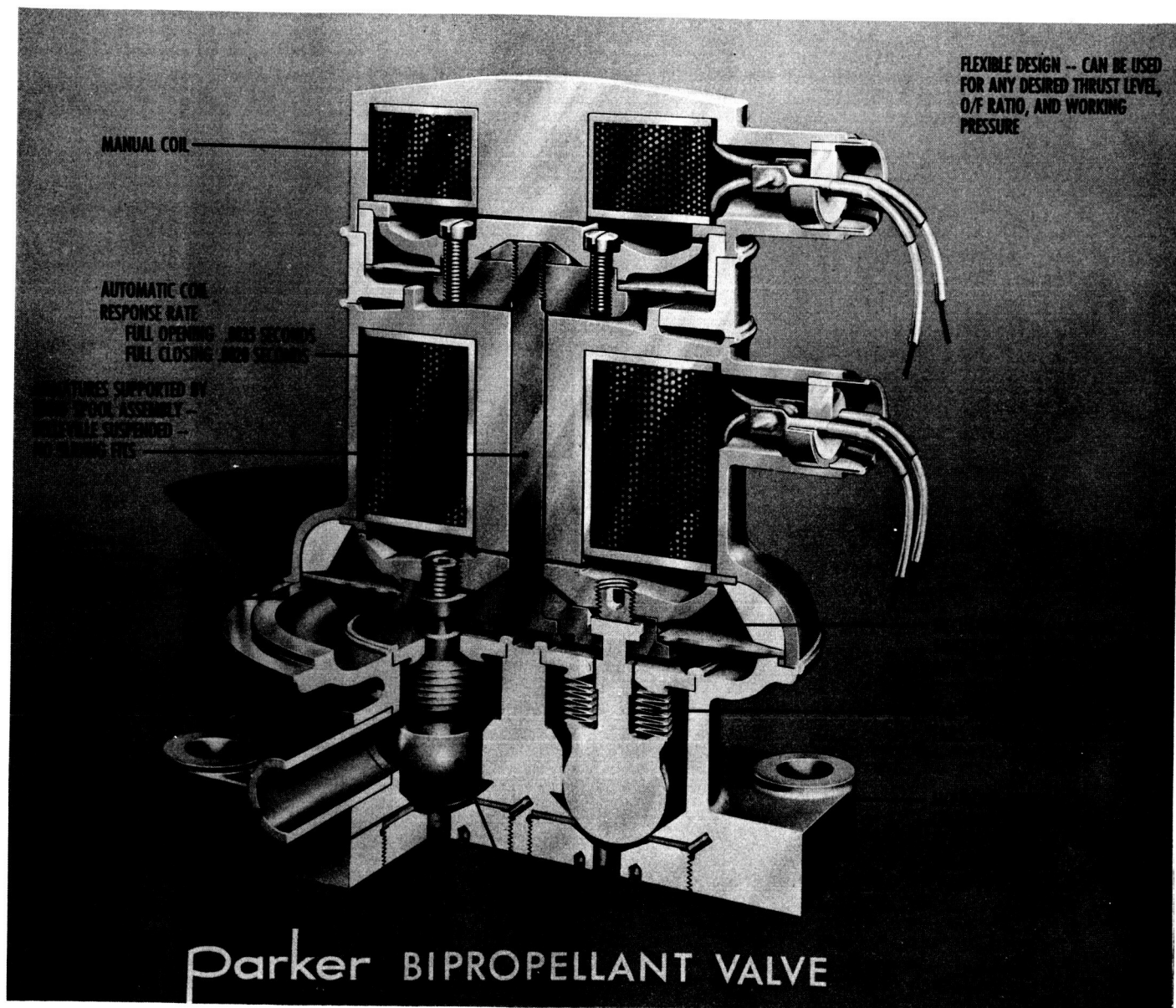


Figure V-165. Parker Bipropellant Valve

Vendor		National Water Lift (Biprop)		National Water Lift (Dual Solenoids)	
Design Item	Max. Possible Rating	Comments	Rating	Comments	Rating
General Design:					
1. Type Actuation	10	Conventional solenoid. Rocking yoke connection to poppets.	8	Two solenoids individually operated. Coaxial approach.	4
2. Envelope	20	3.0 x 1.3 x 1.9 plus 1.0 dia x 2.25 for solenoid.	5	4.6 x 5.0 x 2.0	5
3. Weight	25	1.0 lb (estimated)	25	3.0 lb (estimated)	15
4. Reliability	30	Unknown - valve not yet built.	15	Unknown - valve not yet built.	15
5. Contamination Sensitivity	25	Uses metal seats. Thus highly sensitive. No protection filters.	5	Uses metal seats. Thus highly sensitive. No protection filters.	5
Detail Design:					
6. Valve Operation	30	Not pressure balanced. Metal diaphragm seal. Helical spring keeps valve closed. Sliding piston in actuator assembly not too good. Dry operation possible.	25	Not pressure balanced. Separate solenoid for each side. Not tied together mechanically - simultaneous operation not insured. Coil springs keep valve closed. Dry operation possible.	18
7. Design Simplicity	30	2 stroke adjust points. Indirect operation. 18 main valve and actuator parts.	24	No adjustment points. Direct operation. 16 main valve and actuator parts.	26
8. Response	25	≈ 0.005 sec to open or to close.	24	≈ 0.005 sec to open or to close.	24
9. Power Requirements	20	≈ 80 watts (28 vdc) primary. ≈ 28 watts (28 vdc) secondary.	18	112 watts (28 vdc) primary. 56 watts (28 vdc) secondary.	15
10. Pressure Drop	20	Not important since a drop of 50-75 psi is required.	18	Not important.	18
11. Seat Leakage	25	Flexible Ball Poppet. Claim < 5 scch GN ₂ but is doubted.	18	Flexible Ball Poppet. Claim < 5 scch GN ₂ but is doubted.	18

BELL AEROSYSTEMS COMPANY

MOOG Servocontrols		Parker Aircraft		Vickers	
Comments	Rating	Comments	Rating	Comments	Rating
Torque motor - specialized materials. Direct angular movement. Requires use of current limiter.	9	Solenoid - specialized materials - large area. Direct linear motion. Several poppet adjustments.	10	Conventional solenoid. Rocking yoke connection to poppets. Can have bad tolerance build-ups.	6
2.0 x 3.7 x 2.9 plus outlet manifold.	18	2.0 dia x 2.88 long plus outlet manifold and inlet tubes.	20	4.0 dia x 2.4 long	15
1.5 lb (estimated)	22	1.2 lb (estimated)	24	4.0 lb (estimated)	10
2,000,000 cycles on flexure sleeves. 700,000 cycles on valve.	30	750,000 cycles on bellows. 100,000 cycles on valve.	25	Unknown - Valve not yet built. Another bellows type valve tested to 250,000 cycles.	20
Fairly small seat with 0.022 stroke. Salts can be collected by sleeve. Not as bad as bellows.	25	Large seat with 0.012 stroke not sensitive to contamination. Folds in bellows collect oxidizer salts producing large particles.	22	Bellows can collect salts making large particles. Large seat with 0.010 stroke is good.	20
Not pressure balanced. Flexure sleeve isolation. Permanent magnet and flexure spring rate keep valve closed. Dry operation possible.	30	Pressure balanced bellows assembly. Requires pressure to open fully. Positive dry checkout such as on "pad" cannot be done. Always need pressure.	28	Double pressure balancing bellows. Relies on pressure and bellows rate to close. Bellows critical for balancing. Dry operation possible.	20
2 stroke adjustment points. Direct operation. 15 main valve and actuator parts.	30	5 stroke adjustment points. Bellows assembly is critical dimensionally. 20 main valve and actuator parts. Direct operation.	26	No adjustment points. Indirect operation. 16 main valve and actuator parts.	25
<0.005 sec to open. <0.005 sec to close.	24	<0.005 sec to open. <0.003 sec to close.	25	≅0.005 sec to open. ≅0.005 sec to close.	24
28 watts (28 vdc) primary. 28 watts (28 vdc) secondary.	20	70 watts (28 vdc) primary. 28 watts (28 vdc) secondary.	19	≅80 watts @ 28 vdc, primary. ≅28 watts @ 28 vdc, secondary.	18
Not important - However drop within ±3 psi can be easily adjusted during assembly.	20	Not important.	18	Not important.	18
Fixed poppet and seat. 0 scch GN ₂ Demonstrated. Never >2.0 scch GN ₂ .	25	Slightly flexible poppet. 2.8 scch GN ₂ demonstrated.	23	Flexible poppet. <5.0 scch GN ₂ estimated.	20

Figure V-166. Propellant Valve Rating

Vendor		National Water Lift (Biprop)		National Water Lift (Dual Solenoids)	
Design Item	Max. Possible Rating	Comments	Rating	Comments	Rating
12. Residual Volume	30	Oxid. Side: 0.005 in. ³ Fuel Side: 0.004 in. ³	28	Oxid. Side: 0.022 in. ³ Fuel Side: 0.022 in. ³	23
13. Material Compatibility	30	Materials not yet selected. Diaphragm material could be problem area.	25	Wetted parts 300 SST, 17-7 and K-601. 17-7 may produce problems in Nitric Acid.	25
14. Dynamic Sealing	25	Metal diaphragm welded to body.	17	None due to coaxial design approach.	25
15. External Sealing	15	Only 2 joints i.e. at diaphragm. Excellent approach.	15	Welded and plastic seals. 6 joints.	12
16. Filtration	20	Filter not incorporated.	0	Filter not incorporated.	0
17. Structural Aspects	20	Fair - Actuating yoke may yield vibration problems.	15	Appears to be very good.	18
18. Environmental Resistance	15	Assumed to be good.	15	Might have problem with solenoid shell material.	12
19. Repair and Maintenance	5	Coils easily replaceable. Poppet repair difficult.	4	Poppet repair easy. Coils welded shut.	4
20. Miscellaneous	5	No comment.	5	No comment.	5
21. Vendor's Experience	20	No experience in this type valve. General valve experience very good.	20	Considerable experience in coaxial solenoid valves.	25
22. Bell Confidence in Vendor	30	Good Confidence. Several projects completed for Bell.	22	Good Confidence. Several projects completed for Bell.	22
Total Rating	485		351		340

MOOG Servocontrol		Parker Aircraft		Vickers	
Comments	Rating	Comments	Rating	Comments	Rating
Oxid. Side: 0.0079 in. ³ Fuel Side: 0.0032 in. ³	27	Oxid. Side: 0.0026 in. ³ Fuel Side: 0.0025 in. ³	30	Oxid. Side: 0.034 in. ³ Fuel Side: 0.033 in. ³	20
Anticipate problems with 17-7 Flexure sleeve in Nitric Acid. Body also 17-7.	25	Anticipate problems with AM355 bellows in Nitric Acid. Remainder of valve good.	23	Valve made entirely of AM350 and 17-7. Will have problems with Nitric Acid.	23
Flexure sleeves pressed into body.	15	Welded bellows. Welded to body.	18	Formed bellows. Welded to body.	17
Welded and metallic seals. 10 joints.	12	All welded. 8 joints	14	All welded. 12 joints	13
Filters not incorporated.	0	Filters not incorporated	0	Filters incorporated 20 nominal, 35 absolute. sst cloth - plain tube.	18
Appears to be excellent.	20	Many adjustment points could cause vibration pro- blems - otherwise good.	15	Appears marginal with flexible poppet assy.	5
No problems anticipated.	15	Proprietary external material might have to be protected by some means to prevent rust- ing.	12	Assumed to be good.	15
Coils easily replaceable. Poppet repair easy by unbolt- ing outlet manifold.	5	1 weldment cut to replace either coil. Poppet repair very difficult.	2	Solenoid easily replaceable, poppet repair extremely difficult.	4
Coil hermetically sealed. Motor case is not. However, will be done by welding motor case to body.	4	Coil temperature compen- sated.	5	No comment.	5
Excellent experience. Almost identical valve built and test- ed. Also hot fired.	30	Excellent experience. Almost identical valve built and tested.	28	Has built small valves of this type but not this par- ticular design. Good experience.	22
Very high confidence. Has completed several develop- ment projects in excelent order for BAC.	30	High confidence based on NASA information. No actual work contract.	27	Have not previosuly worked with them.	
436		414		360	

Figure V-167. Propellant Valve Rating

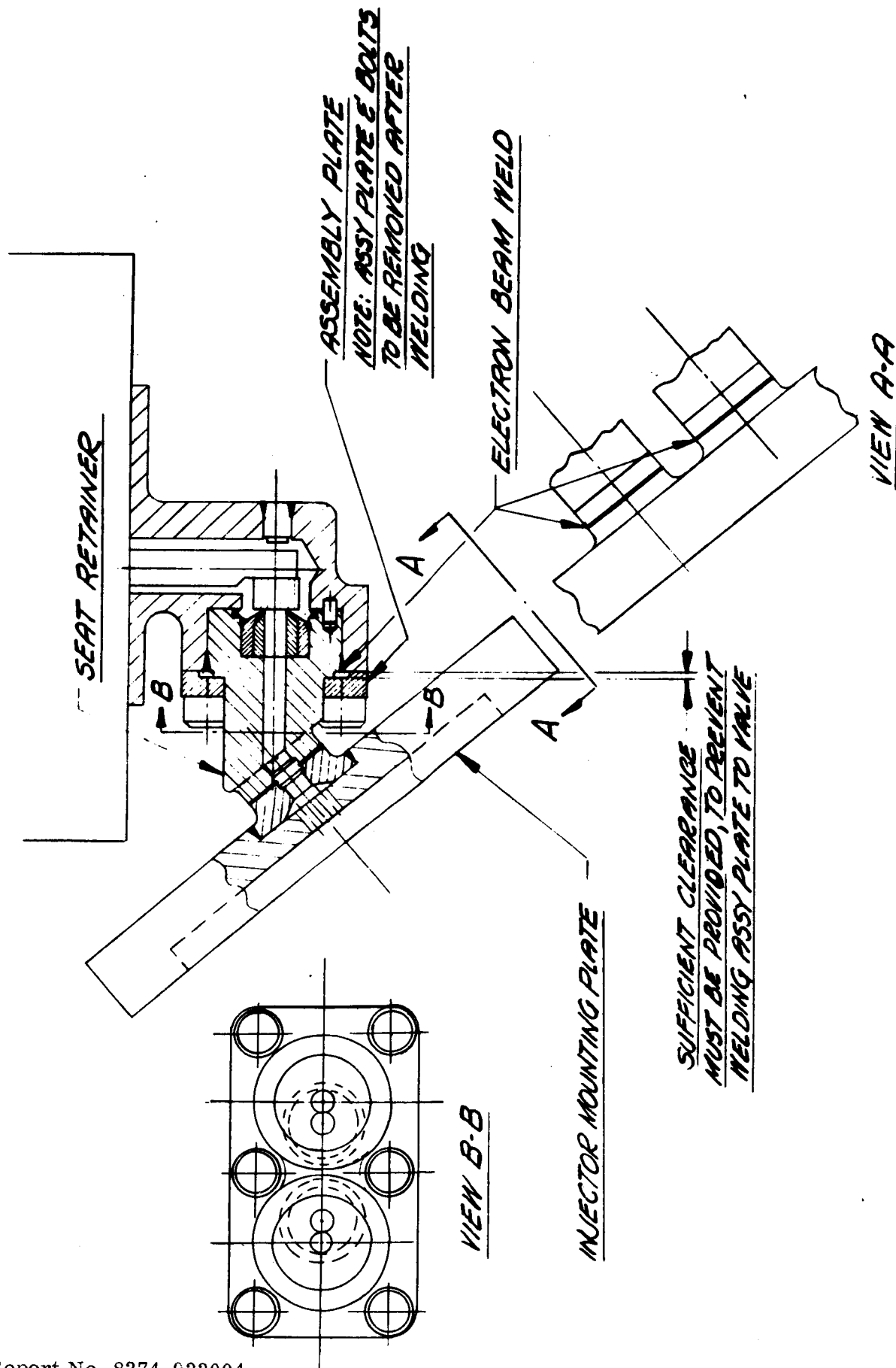


Figure V-168. Layout Showing Welded Interface Connection of Moog Valve to Thrust Chamber

Thrust Range (lb)	Mfgr. & Model No.	Size (in.)	Weight (lb)	Response (sec)	Resid. Vol. (in. ³)(O/F)	Power* (watts)	Leakage (Sec GN ₂ per hr ²)	Actuator Type
5-15	Moog 50-259	2.6 x 2.3 x 2.1	0.58	0.003-0.004	0.0033/0.0021	25	< 5.0	TM
15-50	Moog 52-106	3.2 x 3.0 x 2.2	0.80	0.004-0.005	0.0044/0.0034	25	< 5.0	TM
25-50	Parker	1.5 dia x 2.2	0.70	0.003-0.004	0.004/0.004	60	< 5.0	S
50-100	Moog 50-245	3.2 x 3.4 x 2.3	1.20	0.004-0.006	0.008/0.003	28	< 5.0	TM
50-125	Parker 5630116	2.0 dia x 2.4	0.80	0.003-0.005	0.005/0.005	75	< 5.0	S
100-150	Moog 52-100	3.5 x 3.5 x 2.5	1.75	0.005-0.008	0.012/0.007	38	< 5.0	TM
100-200	Bell 8250-472040	4.3 x 4.4 x 1.9	2.70	0.030-0.050	0.040/0.040	8	< 0.5	H
200-400	Parker	3.0 dia x 2.8	3.0	0.008-0.012	0.027/0.021	150	< 5.0	S
400-1000	Parker			0.015-0.020		60	< 5.0	P
400-1000	Bell	4.8 x 4.4 x 2.0	3.5	0.040-0.070	0.15/0.15	12	< 1.0	H
15-100	Fox Throttle Valve	3.0 x 2.8 x 2.4	1.0	Variable		28	**	H

NOTES:

* Includes transistor switch for Moog valves. Power at 28 Vdc.

** Not designed for shutoff

TM = Torque motor

S = Solenoid

H = Hydraulic actuator

P = Pilot operated

Figure V-169. Thrust Range - Propellant Valves

H. PROTOTYPE ENGINE DESIGN SELECTION

Design efforts were initiated to integrate, into a prototype, demonstration engine design, the results of the various concept analyses, materials investigations, insulation studies and tests, injector evaluations, and valve configuration studies.

1. Design Selection

As a result of the foregoing investigations, the adiabatic wall (insulated) design concept was selected. This concept is most attractive to meet the universal requirements of the program.

The unbalanced injector design has demonstrated the ability to provide a considerable decrease in gas temperature along the chamber wall while maintaining performance in excess of 290 seconds specific impulse. The feasibility of this insulated design concept has been demonstrated in preprototype hardware, revealing driving gas temperature less than 2800° F, thereby being completely compatible with the present state-of-the-art coating systems.

Satisfactory insulation evaluations revealed compatibility between the insulation materials and the chamber material and coating system at expected operating temperatures and above.

The design of the prototype engine consists of a columbium chamber (SCb-291) and nozzle (Cb-1% zirconium) coated internally and externally with the oxidation resistant modified silicide coating (508-C supplied by Sylcor Division of General Telephone and Electronics Co.); a columbium (Cb-1% zirconium) injector of the unbalanced 8 element triplet (TU-6A) type; a composite insulation system consisting of alumina bubbles around the chamber (contained in a columbium (Cb-1% zirconium) shell) also coated internally and externally with the 508-C silicide coating and Dyna-Quartz around the columbium shell; a fiberglass overwrap and a Moog bipropellant valve.

A cutaway drawing of this prototype engine is shown in Figure V-170.

A cross-section of this engine is given in Figure V-171.

2. Design Review

The design review of the prototype engine was held at NASA-JPL on 4 May 1965 with NASA representatives from Washington Headquarters, JPL, MSC and MSFC in attendance. NASA concurred with the insulated columbium chamber design incorporating the unbalanced type injector TU-6A. The fabrication of two engines will commence upon completion of final checking of the drawings.

The drawings for the two prototype engines were completed by June 1, 1965 and a list of these drawings is given below:

<u>Drawing No.</u>	<u>Title</u>
8374-470001	Thrust Chamber Assembly
8374-470002	Chamber, Assembly of
8274-470003	Chamber
8374-470004	Insulation Jacket
8374-470005	Nozzle
8374-470006	Support Strap
8374-470007	End Strap, Assembly of
8374-470008	Shell
8374-470010	Bolt, Valve, Special
8374-470011	Flange, Mounting
8374-470012	Seal Retainer
8374-470013	Ring, Seal Spacer
8374-470014	Plug
8374-473100	Injector, Assembly of
8374-473101	Orifice Plate, Assembly of
8374-473102	Orifice Plate
8374-473103	Injector Blank
8374-473104	Inserts
8374-473105	Insulator
8374-473106	Adapter Plate
8374-473107	Valve Support
8374-473108	Tubes, Propellant Feed
8374-473110	Support Cone
Model 52 x 125	Moog Propellant Valve

Thermal and structural analyses were performed to fully evaluate this design and are described in the following paragraphs. The basic installation conditions agreed upon for this engine were as follows:

- (a) The engine shall radiate to an enclosure held at +120° F.
- (b) The heat conducted from the injector mounting flange to the simulated vehicle shall be 550 BTU/hour with the vehicle support at +400° F.
- (c) The engine shall be supported at the injector end and the nozzle end in the insulated configuration and at the injector end only in the radiation free configuration.

3. Thermal Analyses

Rather extensive thermal analyses were conducted on this design to determine temperature distributions for both steady state and transient heatup and shutdown conditions. A brief description of the computer programs used for these analyses is given in Appendix II.

The prototype design was subdivided into 67 nodal points, as illustrated in Figure V-172, which represents the mathematical model that was used. The boundary conditions imposed on the thermal model were as follows:

Effective gas temperature = 2600° F.

Throat Heat Transfer Coefficient = $0.000836 \frac{\text{BTU}}{\text{in.}^2 \text{sec}^\circ \text{F}}$

$$\frac{h_g}{h_{g^*}} = \left(\frac{A}{A^*} \right)^{-1.2}$$

$$1 \leq \frac{A}{A^*} \leq 4$$

Divergent Nozzle Heat Transfer Coefficient

$$\frac{h_g}{h_{g^*}} = 1.515 \left(\frac{A}{A^*} \right)^{-0.9}$$

$$\frac{A}{A^*} \geq 4$$

Chamber Heat Transfer Coefficient

$$\frac{h_g}{h_{g^*}} = \left(\frac{A}{A^*} \right)^{-0.8}$$

Radiation out exit plane = $\epsilon = 0.7$

(Geometric view factors are governed by configuration-exhaust gases transparent to infrared.)



Pulse mode is achieved by throttling; i.e., on times are extremely short unless otherwise stated. The heat flux through the structural support was that required to maintain it at approximately 400° for steady state. For the transient runs, no heat was given off while the support was below 400° F. Heat is drained away and recorded to maintain the support at 400° F when the support is potentially above 400° F.

Representative steady state temperatures for duty cycles of 1.0, 0.1, and 0.01 are shown on the mathematical model in Figures V-173, V-174, and V-175, respectively.

Tables V-24, V-25, and V-26 are printed out steady state results for duty cycles of 1.0, 0.1, and 0.01, respectively. The node temperatures are presented from 1 to 10 from left to right, and ascending numbers going down; i.e., second row 11 to 20, etc., and the numbers are keyed to the mathematical model. The heat flux out through the head end flange support is given at the upper left of these tables; 550 BTU/hr for duty cycle of 1.0, 390 and 120 for the other two duty cycles. These were not the exact amounts of heat loss to maintain 400° F at element 57, but they were close approximations.

In addition to the temperatures shown in these tables, there are heat flux data shown in the lower part of the table.

For nodes 67, 68, and 69 which represent the valve, the valve cover and the valve drive mechanism, respectively, the net heat flux by conduction to the node is given in the first column. Positive denotes net heat input, negative denotes net heat output. The next column indicates net internal radiation from each node. The next 3 columns have no significance. The "Control Rad (Y)" column indicates the amount of radiation the valve and the valve cover are receiving from the 120° F environment which is very small because the temperature of the valve is not greatly different from 120° F. The "Control Conv (Z)" column indicates the amount of cooling that the propellant provides the valve and the amount of heat dissipated by the valve actuation. All of these heat fluxes are presented in BTU/hr. It can be noted that the overall net heat flux to each element is zero, a rather necessary condition to achieve a heat balance.

TABLE V-24
STEADY STATE TEMPERATURES (DUTY CYCLE = 1.0)

MODEL 8374 100 LB THRUST CHAMBER 80 PSIA									
HC MAX = 433 TITANIUM FLANGE PROP VALVE - ELEM 67									
TG = 2600 DEG F Q OUT = 550									
STEADY STATE TEMPERATURES (DUTY CYCLE = 1.0)									
ITERATION = 6									
() ELEMENT OR NODE NUMBERS									
NETWORK NODE TEMPERATURES DEG. F									
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)
2162.6174	2476.1963	2538.9318	2546.4791	2544.2379	2524.0717	2434.1239	2322.9872	2166.6615	1989.9207
(11)	1870.3807	1752.4234	1591.3189	989.6245	1068.0320	1760.0523	2068.2256	2190.1698	2230.8498
(21)	2134.7137	2061.8806	1955.9387	1027.6912	1490.3419	1792.7863	1954.7050	2022.3181	2020.5983
(31)	1975.2519	1923.6591	473.2486	814.8781	961.6920	1039.6036	1072.0309	1071.2751	1061.8694
(41)	1024.6566	1022.2498	987.6679	954.1624	227.0258	301.2260	329.4640	343.4699	349.1307
(51)	347.5436	345.4026	341.1352	340.9206	371.3260	394.7355	423.4586	787.2145	990.2536
(61)	505.7802	483.4134	892.1451	64.2686	255.7858	642.6492	62.3433	81.0856	249.8542
CONTROL NODE TEMPERATURES DEG. F									
RAD (Y) 120.0000									
CCNV (Z) 2600.0000 -80.0000 -742.0000									
NETWORK NODE HEAT FLOWS BTU / HR									
+ DENOTES HEAT FLOW TO A NODE									
- DENOTES HEAT FLOW FROM A NODE									
CONTROL CONTROL CONTROL CONTROL CONTROL									
ACDE	INT COND	INT RAD	EXT RAD	CONV (W)	CONV (X)	RAD (Y)	CONV (Z)	HEAT FLUX	CONTROL
67	176.7459	1.7372	-0.	0.	0.	1.7422	-180.2249	0.	0.
68	-12.9504	9.8641	-0.	0.	0.	3.0863	0.	0.	0.
69	-162.5963	-11.6013	-0.	0.	0.	0.	174.1976	0.	0.
CONTROL NODE HEAT FLOWS BTU / HR									
+ DENOTES HEAT FLOW TO A CONTROL									
- DENOTES HEAT FLOW FROM A CONTROL									
RAD (Y) 997.0297 1									
CCNV (Z) -21583.9556 1									
HT FLX (C) 550.0000 1									
8207.8903 2 -174.1976 3									

TABLE V-25
STEADY STATE TEMPERATURES (DUTY CYCLE = 0.1)

MODEL 8374	100 LB THRUST CHAMBER	80 PSIA
HC MAX = 433		
TG = 2600 DEG F	TITANIUM FLANGE	PROP VALVE - ELEM 67
Q OUT = 390		
DUTY CYCLE = 0.1		

ITERATION = 7									
NETWORK NODE TEMPERATURES DEG. F									
1406.2685	1770.0213	1966.5858	2001.2139	1989.6425	1930.5744	1842.3154	1713.4871	1512.2173	1276.0176
1125.3862	972.9869	826.5252	629.9115	838.3663	1220.1397	1503.4678	1671.2036	1714.1951	1664.3448
1557.3489	1436.9039	1259.8658	801.7653	1095.4370	1326.8959	1471.7780	1525.7150	1500.1745	1442.0998
1374.6376	1244.5230	389.1329	621.2118	735.4662	806.3014	832.5824	820.2403	791.8271	758.9305
695.1647	649.8129	588.1425	549.3035	204.7758	260.5902	285.9856	300.0618	303.3041	303.0026
297.3868	290.7522	277.4815	267.7707	277.9837	286.6004	391.0404	649.1009	806.2404	1029.3323
430.6168	802.2616	729.9746	88.3004	228.6686	546.6083	65.5127	71.5203	90.8137	
CONTROL NODE TEMPERATURES DEG. F									
RAD (V)	120.0000								
CCNV (Z)	2600.0000	60.0000	762.0000						
NETWORK NODE HEAT FLOWS BTU / HR									
ACDE	INT COND	INT RAD	EXT RAD	CCNV (W)	CONV (X)	RAC (Y)	CONV (Z)	CONTROL	CONTROL
67	40.5850	0.1530	-0.	0.	0.	1.6601	-42.3981	HEAT FLUX	HEAT FLUX
68	-4.4275	0.7046	-0.	0.	0.	3.7229	0.	0.	0.
69	-21.9627	-0.8577	-0.	0.	0.	0.	22.8203	0.	0.
CONTROL NODE HEAT FLOWS BTU / HR									
RAD (V)	614.3686	1							
CCNV (Z)	-5483.1177	1	1507.2620	2	-22.8203	3			
HT FLX (C)	390.0000	1							

Finally, the last three rows are the amounts of heat accepted (positive) and rejected (negative) by each of the control nodes. It can be seen that for the duty cycle of unity, the radiated heat from the outside surface of the engine to the 120°F environment was nearly 1000 BTU/hr; the heat flux to the structure was 550 as stated previously; and the heat input to the chamber from the combustion process was large by comparison, 21,600 BTU/hr. The heat that is rejected to the propellant in the valve and injector is 8200, and the heat added to the system by the valve actuation is 175 BTU/hr. There is an imbalance among these elements because of the radiation off the inner surface of the nozzle out the exit plane. The net radiation out the nozzle exit is 12,000 BTU/hr, or 3.3 BTU/sec, which means a flux of 0.116 BTU/in.²sec at the exit plane.

Several transient temperatures were computed, and a summary of these results are presented in Figures V-176 through V-179. Again transient temperatures were obtained for all 67 nodal points, with those of particular interest being plotted.

Figure V-176 illustrates a cool-down transient from a steady state duty cycle of unity condition. It can be seen that the chamber wall "4" cools monotonically at an ever-decreasing rate. The injector "62" heats up rapidly on shutdown because of the hot chamber wall, and then decays along with the chamber wall temperature. The valve temperature "67" rises slowly with time because it is by design loosely coupled to the thrust chamber. For this analysis the valve was insulated at the propellant inlets so that an actual installation of the unit would be less critical, thermally.

Figures V-177, V-178, and V-179 illustrate representative start transient temperature time histories for the chamber "4", the columbium insulator retainer "28," the injector "62," and the exterior surface "56" for duty cycles of 1.0, 0.1, and 0.01. The chamber heats more slowly for the lower duty cycle, as one would expect, as do the insulator and back face. The injector heats up more quickly for the lower duty cycles because of less cooling being provided by the propellants.

Finally, an actual pulse mode duty cycle for the throat station is shown in Figure V-180 for 0.1 second "on", 0.9 second "off", and this saw tooth curve is superimposed on

the actual throttle type transient for that same station and it tends to illustrate the appropriateness of using the throttle-type duty, thus saving computer time over the actual inputting of on-off cycles.

To further define the characteristics of the prototype engine design, two modifications were made to the original model. The first modification consisted of replacing the alumina bubble insulation with Dyna Quartz insulation, resulting in a complete insulation system of Dyna Quartz. The columbium shell that contained the alumina bubbles was therefore eliminated. For convenience in the modification of this thermal model, the nodes for the columbium shell were not used (nodes 24 through 32).

The second modification to the thermal model consisted of refining the injector analysis. In the original thermal model one node (node 62) represented the average injector temperature. In the modified model the injector was divided into 15 nodal points including 12 elements for the injector face and 3 elements for the capillary tubes. Details of the injector thermal model are shown in Figure V-181.

The face of the injector is divided into two wedge-shaped segments. Segment "A" represents the mass surrounding the injector orifices, and segment "B" represents the mass between the injector orifices. Each segment is divided into a front section and a rear section. The following tabulation identifies the injector nodal points:

<u>Location</u>	<u>Node No.</u>
Segment "A" front	70, 74, 78
Segment "A" rear	71, 75, 79
Segment "B" front	72, 76, 80
Segment "B" rear	73, 77, 81
Center fuel tube	62
Oxidizer tube	82
Outboard fuel tube	83

Note that node 62, which was the single element for the injector in the original thermal model, is now repositioned at the center fuel tube.

The identification and location of the other nodal points on the injector assembly and propellant valve remain the same. They are as follows:

<u>Location</u>	<u>Node No.</u>
Support Cone	59
Injector-Chamber Joint	60
Support Web	61, 63
Valve Flange	64
Valve Body	67
Valve Cover	68
Valve Driving Mechanism	69

Equilibrium temperatures during continuous firing and quasi-steady temperatures during pulse mode operation were determined for each of the nodal points on the modified mathematical model.

Tables V-27 through V-31 are printed out steady state results for duty cycles of 1.0, 0.1, 0.01, 0.001, and 0.0001, respectively. As previously mentioned, the temperature values indicated for nodes 24 through 32 have no significance. Table V-32 provides a summary of the injector temperature profile for the various steady state duty cycles. Transient temperatures during the shutdown following these steady and quasi-steady state conditions were also computed to obtain the maximum valve and injector temperatures which occur during the heat soakback period. For these analyses the propellant valve was insulated at the propellant inlets. In an actual installation the unit would be less critical, thermally.

During the shutdown period, when the injector is uncooled, the temperature gradient across the injector rapidly diminishes. For all the transient analyses the 12 nodal points for the injector were replaced with a single weighted average nodal point, therefore requiring considerably less digital computer time.

The transient shutdown temperatures for representative nodal points for the chamber, the injector and the propellant valve, are presented in Figures V-182 through V-185.

Figure V-182 illustrates the temperature time histories for these nodal points following a steady state firing (duty cycle = 1.0). Although the propellant valve had not reached its peak temperature by the end of 1500 seconds, a quadratic curve was used to obtain the maximum temperature of 237°F. Figures V-183, V-184, and V-185 similarly illustrate the shutdown transients for these nodes following duty cycles of 0.1, 0.01, and 0.001, respectively.

TABLE V-27
STEADY STATE TEMPERATURES (DUTY CYCLE = 1.0)

MODEL 8374		100 LB THRUST CHAMBER		80 PSIA	
PG MAX = 433		TITANIUM FLANGE		PRCP VALVE - ELEM 67	
TG = 2600 DEG F		INJECTOR - ELEMENTS 70 THRU 81			
		STEADY STATE TEMPERATURES (DUTY CYCLE = 1.0)			
		ALUMINA BUBBLES REPLACED BY DYNA QUARTZ			
() Element or Node Numbers					
ITERATION = 7					
NETWORK NODE TEMPERATURES DEG. F					
(1)	(2)	(3)	(4)	(5)	(6)
2161.2431	2490.4738	2553.1646	2557.8896	2552.7782	2530.4962
1868.2502	1752.1785	1591.2865	589.6122	489.8291	1480.1563
1605.1249	1697.3162	1785.8071	-0.0019	-0.0020	-0.0020
-0.0020	-0.0020	197.3361	572.0949	651.4534	665.0848
803.2906	1020.9587	987.5422	554.1458	148.7725	247.9139
262.7599	276.7826	299.9171	340.3042	371.2564	394.7315
457.7386	62.0402	845.9175	63.8015	164.5056	549.8969
356.1937	393.0851	360.4886	390.3056	348.2511	489.8627
657.3246	60.4849	60.9127			
CONTROL NODE TEMPERATURES DEG. F					
CCND (W)	0.	400.0000			
RAD (Y)	120.0000				
CCNV (Z)	2600.0000	60.0000	762.0000		
NETWORK NODE HEAT FLOWS BTU / HR					
+ DENOTES HEAT FLOW TO A NODE - DENOTES HEAT FLOW FROM A NODE					
ACCE	INT CCND	INT RAD	EXT RAD	CCND (W)	CCNV (Z)
67	176.4585	1.7372	-0.	0.	1.7423
68	-12.9505	9.8640	-0.	0.	3.0865
69	-162.5972	-11.6012	-0.	0.	0.
CONTROL NODE HEAT FLOWS BTU / HR					
+ DENOTES HEAT FLOW TO A CONTROL - DENOTES HEAT FLOW FROM A CONTROL					
CCND (W)	374.3490	1			
RAT (Y)	725.5478	1			
CCNV (Z)	-21085.3597	1	8123.9363	2	-174.1585
					3

TABLE V-28
STEADY STATE TEMPERATURES (DUTY CYCLE = 0.1)

MODEL 8374 100 LB THRUST CHAMBER									
PG MAX = 433									
TG = 2600 DEG F									
INJECTOR - ELEMENTS 70 THRU 81									
DUTY CYCLE = 0.1									
ALUMINA BUBBLES REPLACED BY DYNA QUARTZ									
ITERATION = 7									
80 PSIA									
PRCP VALVE - ELEM 67									
TITANIUM FLANGE									
DEG. F									
NETWK	NODE	TEMPERATURES	DEG. F	(1)	(2)	(3)	(4)	(5)	(6)
1412.7097	1816.1044	2032.5421	2063.2644	2045.3940	1999.4417	1883.7368	1748.1125	1534.8820	1275.1650
1111.6795	968.5331	824.8550	628.9317	413.9693	984.2710	1272.6150	1418.6295	1428.5144	1257.5428
1227.0316	1209.7938	1152.7054	-0.0019	-0.0020	-0.0020	-0.0020	-0.0020	-0.0020	-0.0020
-0.0020	-0.0020	182.5092	413.1378	507.1896	554.0232	557.0058	487.1862	501.8530	522.2585
552.8447	642.6281	585.8010	548.3654	143.1928	209.2413	233.8192	245.2306	245.7479	230.0539
233.1367	237.9840	245.5175	265.8251	277.3444	286.3054	400.3837	577.1124	753.0534	1002.9345
403.5975	101.2553	703.4660	86.0905	157.6288	506.6942	65.3494	71.7705	90.6576	715.4168
718.6217	719.9117	719.6030	735.3080	729.1028	755.5880	748.2888	831.6497	811.1865	854.9745
834.1942	71.2053	73.6642							
CONTROL NODE TEMPERATURES DEG. F									
CCND (W)	0.	400.0000							
RAC (Y)	120.0000								
CCNV (Z)	2600.0000	60.0000	762.0000						
NETWK NODE HEAT FLOWS BTU / HR									
+ DENOTES HEAT FLOW TO A NODE - DENOTES HEAT FLOW FROM A NODE									
NLDE	INT CCND	INT RAC	EXT RAC	CCND (W)	CCNV (X)	RAC (Y)	CCNV (Z)	CONTROL	HEAT FLUX
67	35.3257	0.1529	-0.	0.	0.	1.6644	-41.1423	0.	0.
68	-4.4368	0.7038	-0.	0.	0.	3.7330	0.	0.	0.
69	-21.9689	-0.8567	-0.	0.	0.	0.	22.8256	0.	0.
CONTROL NODE HEAT FLOWS BTU / HR									
+ DENOTES HEAT FLOW TO A CONTROL - DENOTES HEAT FLOW FROM A CONTROL									
CCND (W)	266.4460	1							
RAC (Y)	428.5718	1							
CCNV (Z)	-5316.7539	1	1594.8874	2	-22.8256	3			

TABLE V-29
STEADY STATE TEMPERATURES (DUTY CYCLE = 0.01)

MODEL 8374 100 LB THRUST CHAMBER									
PG MAX = 433									
PG = 2600 DEG F									
INJECTOR - ELEMENTS 70 THRU 81									
DUTY CYCLE = 0.01									
ALUMINA BUBBLES REPLACED BY DYNA QUARTZ									
ITERATION = 7									
NET WORK NODE TEMPERATURES DEG. F									
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)
753.3164	867.4093	946.7049	561.2665	955.8502	941.4447	906.8384	861.5468	775.9196	635.6760
529.8262	432.4587	360.1365	310.1153	303.0857	544.5226	630.3761	684.2475	689.7203	621.7129
624.5030	627.0437	583.9726	-0.0019	-0.0020	-0.0020	-0.0020	-0.0020	-0.0020	-0.0020
-0.0020	-0.0020	160.8051	267.1219	296.5496	314.6150	316.3147	286.6843	298.9371	312.1833
319.7441	341.6468	297.6701	273.8610	135.1595	167.5461	177.3240	182.6013	182.9535	175.1240
178.2980	181.9983	184.3717	190.4766	189.0395	188.7970	400.1589	473.8125	552.8780	665.1278
342.6512	250.7227	503.8895	150.9961	147.3438	444.2025	103.6031	105.4277	106.1180	630.1048
629.0088	630.5049	629.1360	633.0596	630.6505	634.5976	633.3488	644.7456	641.2730	647.0936
644.3820	142.6631	149.7741							
CONTROL NODE TEMPERATURES DEG. F									
CCND (W)	0.	400.0000							
RAC (Y)	120.0000								
CCNV (Z)	2600.0000	60.0000	762.0000						
NET WORK NODE HEAT FLOWS BTU / HR									
+ DENOTES HEAT FLOW TO A NODE									
- DENOTES HEAT FLOW FROM A NODE									
ACDE	INT CCND	INT RAC	EXT RAC	CCND (W)	CCNV (X)	RAC (Y)	CCNV (Z)	CONTROL	HEAT FLUX
67	32.9657	0.0176	-0.	0.	0.	0.5518	-33.5350	0.	0.
68	-1.2607	0.0294	-0.	0.	0.	1.2314	0.	0.	0.
69	-2.1830	-0.0470	-0.	0.	0.	0.	2.2300	0.	0.
CONTROL NODE HEAT FLOWS BTU / HR									
+ DENOTES HEAT FLOW TO A CONTROL									
- DENOTES HEAT FLOW FROM A CONTROL									
CCND (W)	110.3295	1							
RAC (Y)	175.9705	1							
CCNV (Z)	-985.5137	1	229.6408	2	-2.2300	3			

TABLE V-30
STEADY STATE TEMPERATURES (DUTY CYCLE = 0.001)

MODEL 8374 100 LB THRUST CHAMBER 80 PSIA

HG MAX = 433

TG = 2600 DEG F TITANIUM FLANGE PROP VALVE - ELEM 67

INJECTOR - ELEMENTS 70 THRU 81

DUTY CYCLE = 0.001

ALUMINA BUBBLES REPLACED BY DYNA QUARTZ

ITERATION = 7

NETWORK		TEMPERATURES DEG. F									
(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)	(10)	(20)	(30)
277.7913	284.5325	284.8491	279.4266	276.1414	271.7508	263.9342	253.9975	234.3662	199.8166		
169.0752	140.0091	118.3378	105.1966	171.6725	226.2025	232.8143	233.0398	228.2001	213.2340		
211.8238	208.7277	191.4096	-0.0019	-0.0020	-0.0020	-0.0020	-0.0020	-0.0020	-0.0020		
-0.0020	-0.0020	132.6552	157.4317	159.9767	160.0738	158.3329	151.8360	153.4055	154.5043		
151.5308	147.1532	131.6858	116.5584	124.7619	132.8803	133.9630	134.0140	133.3799	131.3423		
131.7585	132.0846	131.0997	129.5425	124.9778	118.3898	259.3881	259.7589	264.3859	270.9549		
216.2259	209.3150	243.5904	172.2151	130.0312	259.3881	154.1929	150.3296	154.1910	268.4977		
268.2224	268.5142	268.2338	268.7259	268.3217	268.8322	268.5877	269.5220	269.1462	269.6891		
269.4572	183.2638	186.0812									

CONTROL NODE TEMPERATURES DEG. F

CCND (W) 0.

RAD (Y) 120.0000

CCNV (Z) 2600.0000 60.0000 762.0000

NETWORK NODE HEAT FLOWS BTU / HR + DENOTES HEAT FLOW TO A NODE
- DENOTES HEAT FLOW FROM A NODE

NODE	INT COND	INT RAD	EXT RAD	CONTROL		CONTROL	RAD (Y)	CONTROL		CONTROL	HEAT FLUX
				COND (W)	CONV (X)			CONV (Y)	CONV (Z)		
67	8.5554	-0.0000	-0.	0.	0.	0.	-1.3111	-7.2443	0.	0.	0.
68	2.6694	0.2083	-0.	0.	0.	0.	-2.8776	0.	0.	0.	0.
69	0.0016	-0.2082	-0.	0.	0.	0.	0.	0.2067	0.	0.	0.

CONTROL NODE HEAT FLOWS BTU / HR + DENOTES HEAT FLOW TO A CONTROL
- DENOTES HEAT FLOW FROM A CONTROL

RAD (Y)	28.9471	1									
CCNV (Z)	-128.1663	1	22.0786	2	-0.2067	3					

STEADY STATE TEMPERATURES (DUTY CYCLE = 0.0001)

MODEL 8374 100 LB THRUST CHAMBER 80 PSIA									
TG = 2600 DEG F HG MAX = 433									
INJECTOR - ELEMENTS 70 THRU 81									
DUTY CYCLE = .0001									
ALUMINA BUBBLES REPLACED BY DYNA QUARTZ									
I T E R A T I O N = 7									
N E T W O R K N O D E T E M P E R A T U R E S D E G . F									
(1) (2) (3) (4) (5) (6) (7) (8) (9) (10)									
97.1396 94.9915 90.9953 87.2026 85.7797 84.1893 81.8943 79.2013 74.0062 65.0371									
57.0091 48.5726 40.5186 34.2291 112.3521 104.5872 102.8182 100.0750 97.4877 97.7214 (20)									
92.7375 84.2547 70.7789 -0.0019 -0.0020 -0.0020 -0.0020 -0.0020 -0.0020 -0.0020 (30)									
-0.0020 -0.0020 118.0916 114.5153 113.8410 112.8624 111.9519 112.3074 109.9372 105.9293 (40)									
98.0147 84.7741 77.7184 63.1175 119.2749 118.0596 117.7777 117.4293 117.1178 117.1759 (50)									
116.3106 114.8910 111.9685 107.0785 101.2079 91.1413 98.6196 98.5627 98.2259 97.7477 (60)									
100.9459 100.3762 99.3468 103.4311 118.4501 98.6196 105.4678 106.9791 105.5624 97.9106 (70)									
97.9113 97.9108 97.9112 97.8917 97.8924 97.8920 97.8894 97.8460 97.8564 97.8450 (80)									
97.8524 99.4767 99.5582									
C O N T R O L N O D E T E M P E R A T U R E S D E G . F									
C O N D (W) 0.									
R A D (Y) 120.0000									
C O N V (Z) 2600.0000 60.0000 762.0000									
N E T W O R K N O D E H E A T F L O W S B T U / H R									
+ DENOTES HEAT FLOW TO A NODE - DENOTES HEAT FLOW FROM A NODE									
N O D E I N T C O N D I N T R A D E X T R A D C O N D (W) C O N T R O L C O N V (X) C O N T R O L R A D (Y) C O N T R O L C O N V (Z) C O N T R O L H E A T F L U X									
67 -0.1424 0.0007 -0. 0. 0. 0.4914 -0.3497 0.									
68 -1.0443 -0.0605 -0. 0. 0. 1.1047 0. 0.									
69 -0.0821 0.0598 -0. 0. 0. 0. 0.0223 0.									
C O N T R O L N O D E H E A T F L O W S B T U / H R									
+ DENOTES HEAT FLOW TO A CONTROL - DENOTES HEAT FLOW FROM A CONTROL									
R A D (Y) -24.2438 1									
C O N V (Z) -13.7313 1 0.7822 2 -0.0223 3									



TABLE V-32
INJECTOR TEMPERATURE PROFILE STEADY-STATE EQUILIBRIUM CONDITION

Nodal Point	Location	Temperature (° F)				
		Duty Cycle				
		1.0	0.1	0.01	0.001	0.0001
59	Support Cone	890	753	553	264	98
60	Joint	1234	1003	665	271	
61	Support Web	458	404	343	216	
70	Injector Face	372	715	630	269	
71	Injector Face	356	719	629		
72	Injector Face	393	720	630		
73	Injector Face	360	720	629		
74	Injector Face	390	735	633		
75	Injector Face	348	729	631		
76	Injector Face	490	756	635		
77	Injector Face	431	748	633		
78	Injector Face	627	832	645		
79	Injector Face	559	811	641		
80	Injector Face	780	855	647		
81	Injector Face	697	834	644		
	Weighted Average of Injector Face Temperature	482	790	640	269	98
62	Center Fuel Tube	62	101	250	209	100
82	Oxidizer Tube	60	71	142	183	99
83	Outboard Fuel Tube	61	73	149	186	99

TABLE V-33
THERMAL ANALYSIS SUMMARY

Conditions	Temperature (°F)				
	Duty Cycle				
	1.0	0.10	0.01	0.001	0.0001
1. With Alumina Bubbles and Dyna Quartz					
a. Injector Average Temp. During Operation	483	802	669	-	-
b. Max. Injector Temp. During Coast	1418	-	-	-	-
c. Valve Temp. During Operation	62	65	105		
d. Valve Temp. 1500 sec after Shutdown	252	-	-	-	-
e. Max. Valve Temp. During Coast (Extrapolated)	276	-	-	-	-
2. With Dyna Quartz Only					
a. Injector Average Temp. During Operation	482	790	640	269	98
b. Max. Injector Temp. During Coast	1308	1058	651	269	98
c. Valve Temp During Operation	62	65	104	154	105
d. Valve Temp 1500 sec After Shutdown	223	211	194	167	105
e. Max. Valve Temp During Coast (Extrapolated)	237	227	215	167	105

Figure V-186 shows the injector temperature as a function of the duty cycle for both types of insulation systems. Both the operating temperature and the maximum expected temperature during shutdown are presented. The operating temperature for the various duty cycles is approximately the same for either the alumina bubble and Dyna Quartz insulation system, or the Dyna Quartz system only. Figure V-187 illustrates the same type information for the propellant valve.

Results of these analyses indicate that when both firing and shutdown are considered, a duty cycle of 1.0 is worse than any pulse mode operation. The maximum valve and injector temperatures following the steady state firing are higher than those of lower duty cycles. Figure V-187 also indicates that the maximum valve temperature is not greatly dependent upon the duty cycle.

A summary of the injector and propellant valve temperatures for the various conditions is given in Table V-33.

4. Structural Analyses

The prototype engine design has been analyzed for structural inputs similar to the Apollo requirements. The structural design considerations are as follows:

<u>Pressure</u>	<u>Operating</u>	<u>Proof</u>	<u>Burst</u>
Max. Ch. Operating (psi)	90	135	225
Max. Valve Operating (psi)	300	450	600
Max. Chamber Temp. °F	3000		
Max. Chamber Stress (psi)	965		
Allow stress at 0.2% creep for one hour (psi) =	1600		
Max. burst pressure of ch. at room temp (psi) = (60,000 psi min. tensile strength)	5200		

Vibration

Calc. nat. freq. = 424 cps

Calc. static g load = 305 g

Long. vibration can be reacted by injector end mount

Lateral vibration reacted by mount at injector end and nozzle end

Thermal

Insulation retainer designed to avoid thermal restraint on chamber

Nozzle and support designed to snub vibrational amplitudes without thermal restraint in long. axis.

The calculated chamber stress for one hour operation is 965 psi. This stress level is well below the allowable stress of 2200 psi for 0.2% creep for one hour at 2800°F, as shown in Figure V-188.

5. Weight Analysis

The estimated weight for the prototype engine and an optimized version is given below:

ENGINE WEIGHT SUMMARY (POUNDS)

Prototype $P_c = 80$, 100% Bell

	Alumina Bubbles and Dyna Quartz	Dyna Quartz	Radiation Version
Propellant Valve	1.25	1.25	1.25
Injector	0.34	0.34	0.34
Supports (Injector and Valve)	1.20	1.20	1.20
Chamber (Less Extension)	1.82	1.82	1.82
Nozzle Extension	2.21	2.21	2.21
Insulation.	7.26	4.48	-
Total	14.08	11.30	6.82
Optimized Version			
$P_c = 100$ psia, RAO Nozzle		8.80	5.20
Actual Weight	15.0		



EXPERIMENTAL AUXILIARY ROCKET ENGINE

CONTRACT NAS 7-305

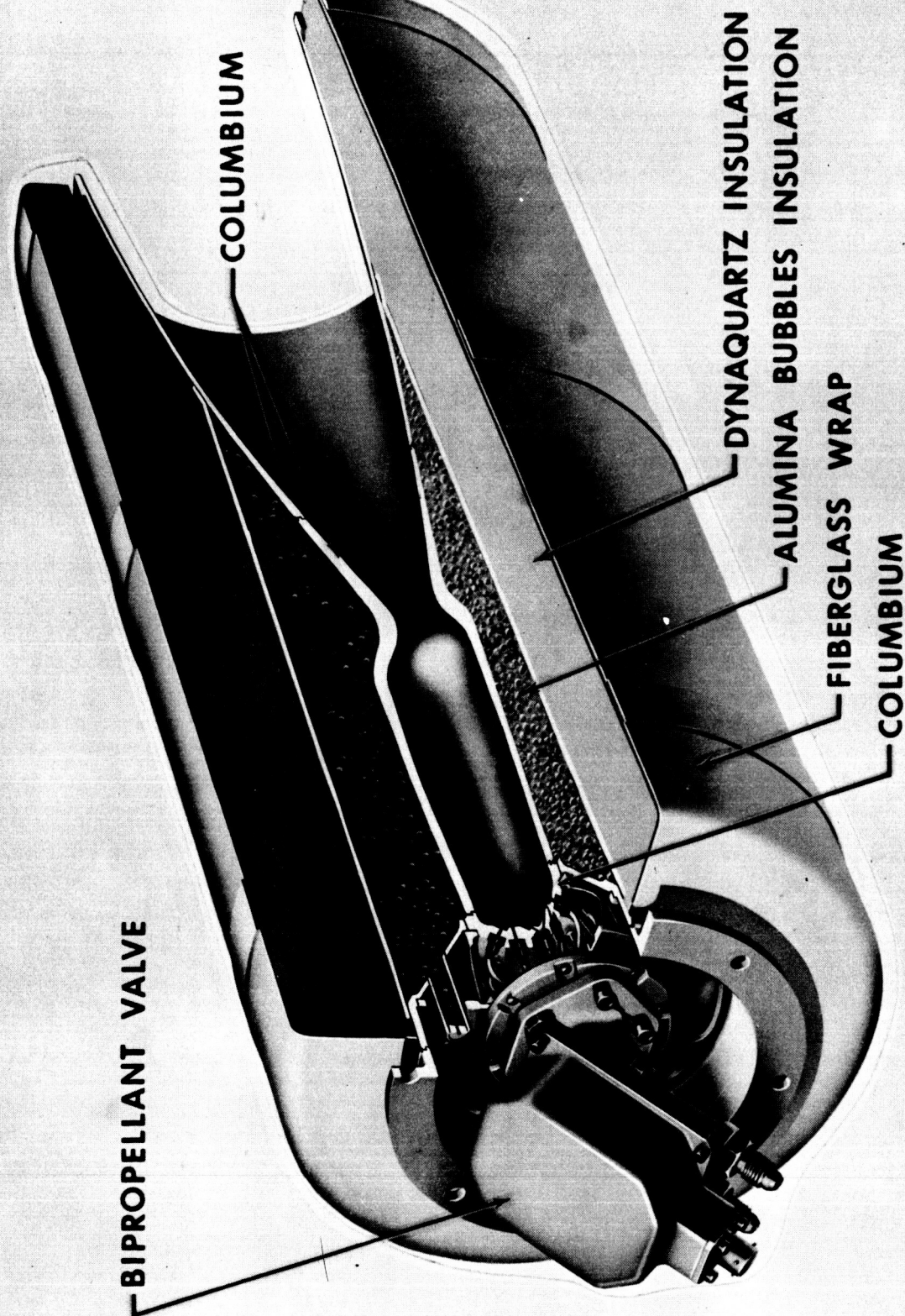
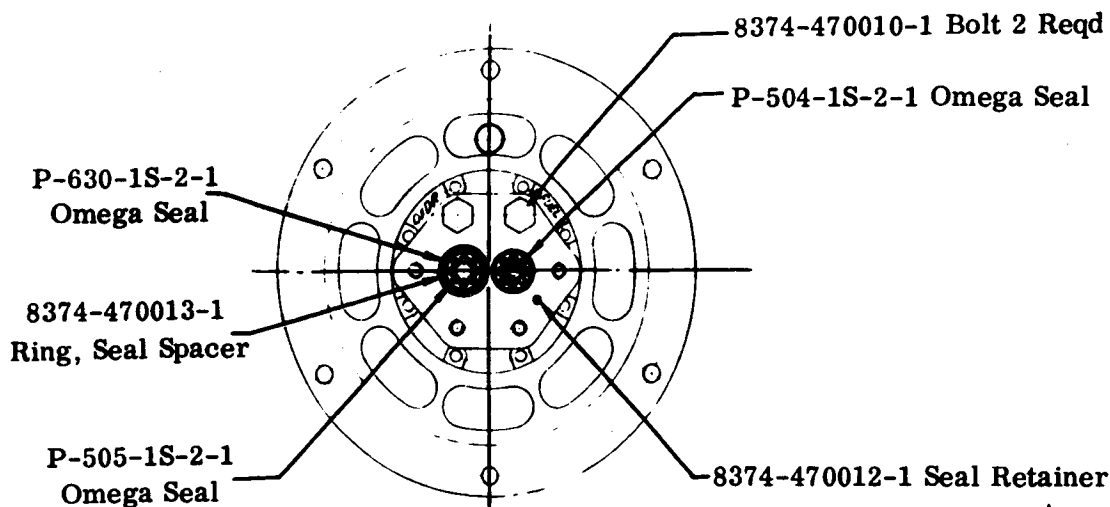
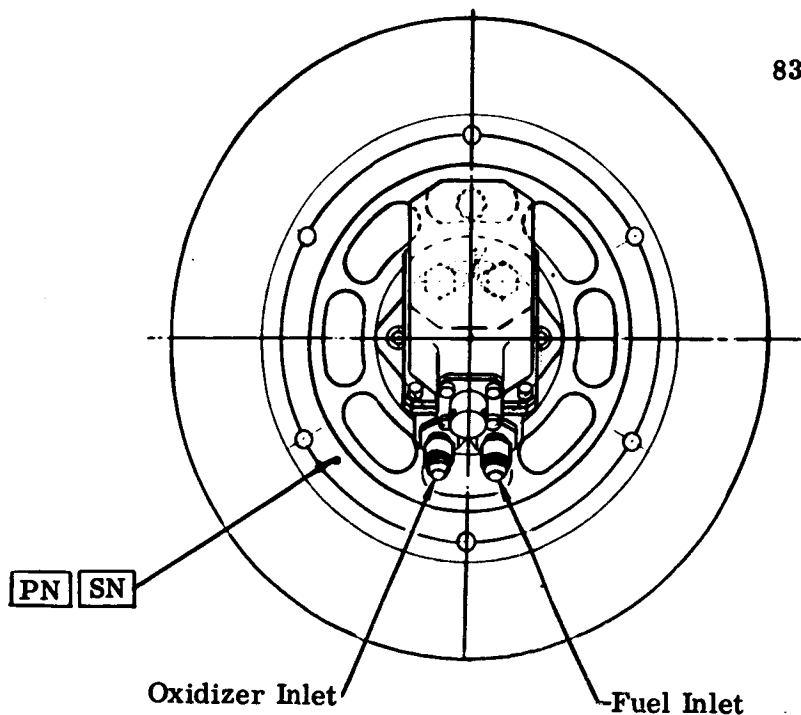


Figure V-170. Prototype Engine



Typical End View

Showing Seal Retainer, Seals, Bolts & Spacer Ring With Valve Remove

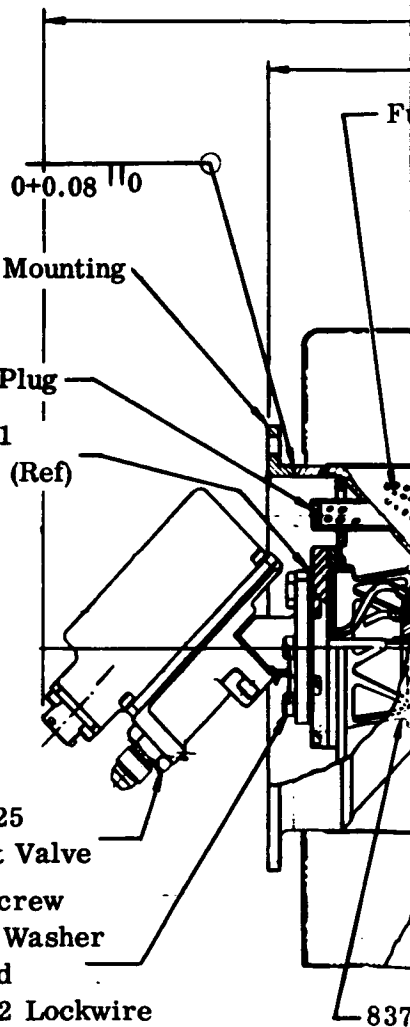


8374-470011-1 Flange Mounting

8374-470014-1 Plug

8374-470012-1 Seal Retainer (Ref)

Model 52 x 125
Bi-propellant Valve
6Z2A832-6 Screw
NAS 620C8L Washer
4 Each Reqd
MS 20995 C32 Lockwire
2 Places



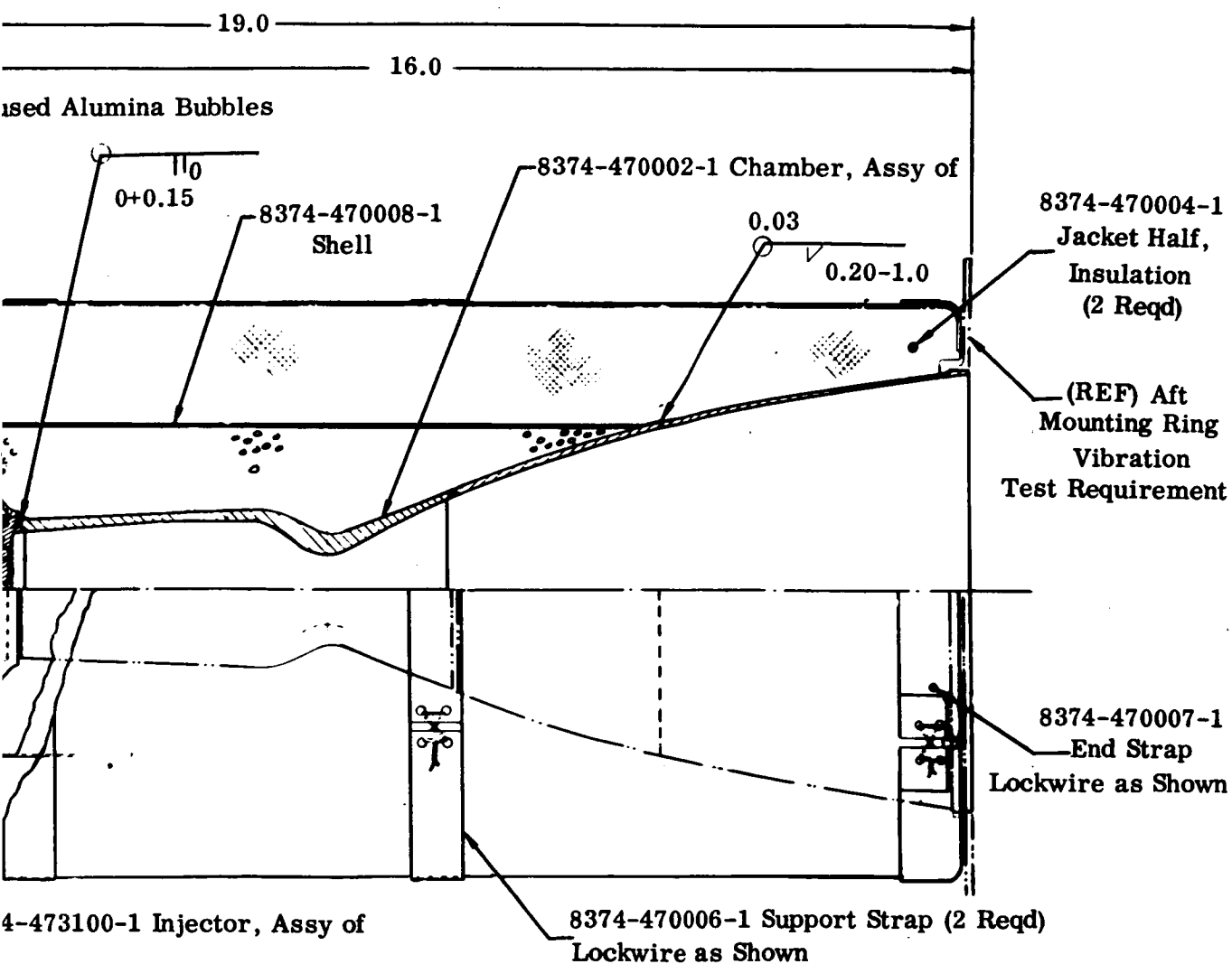


Figure V-171. 8374 Thrust Chamber Assembly

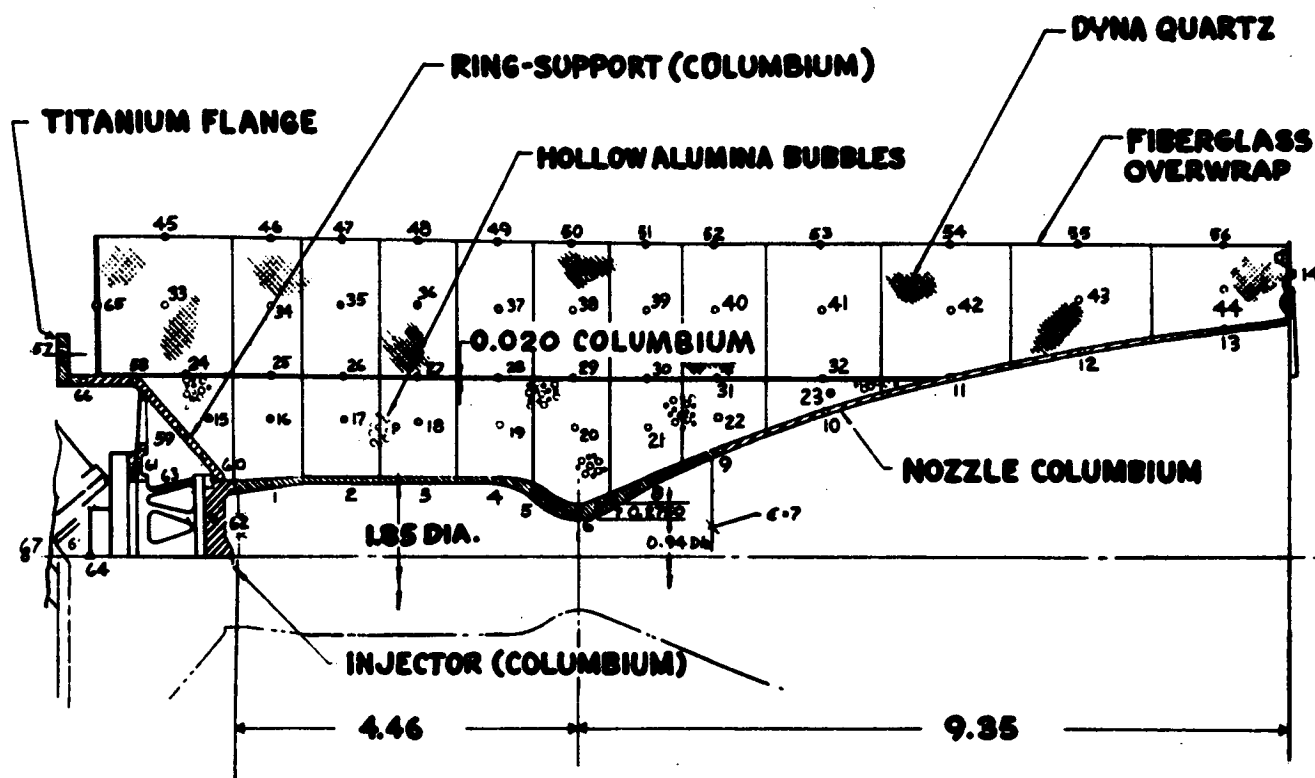
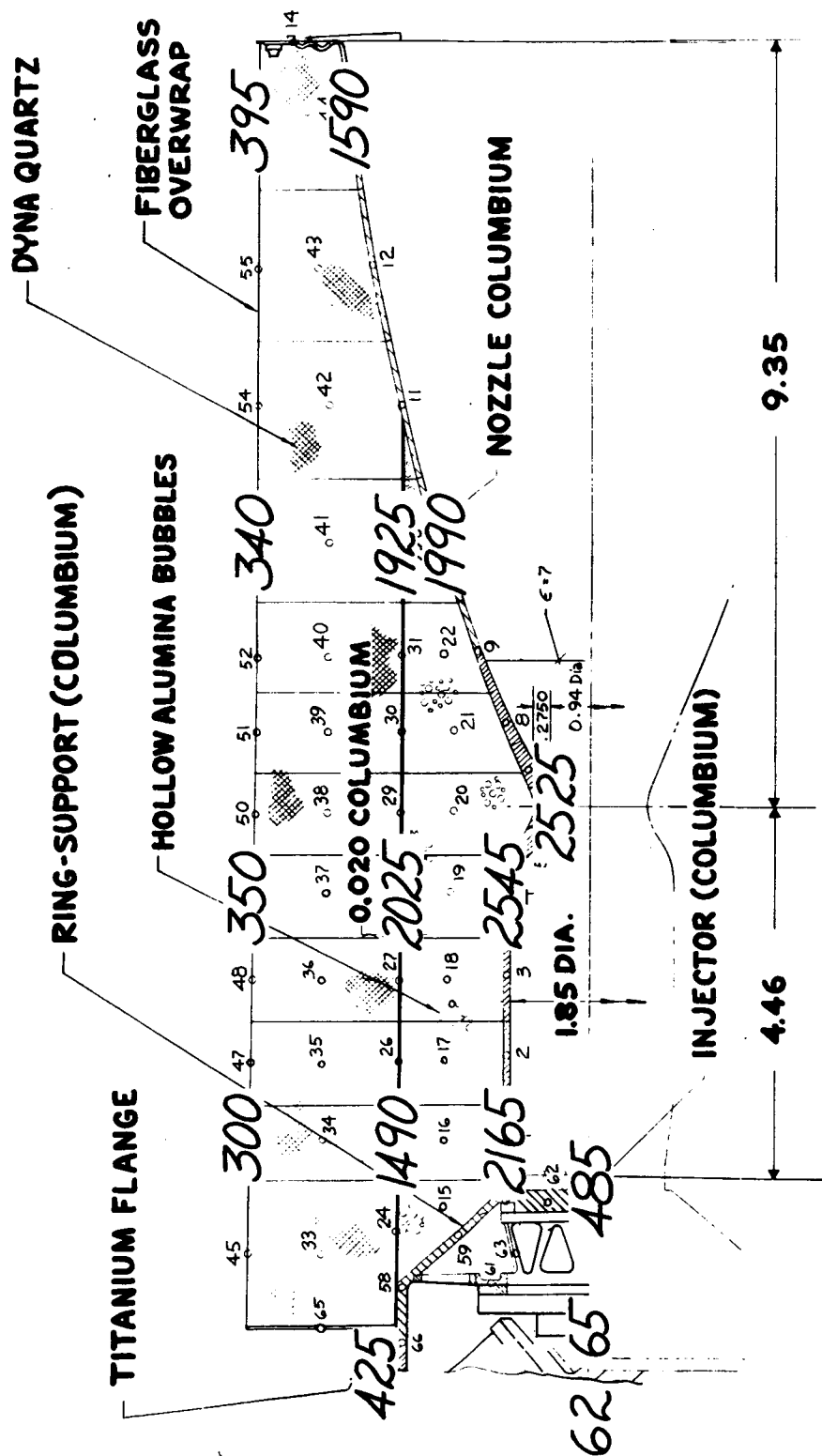
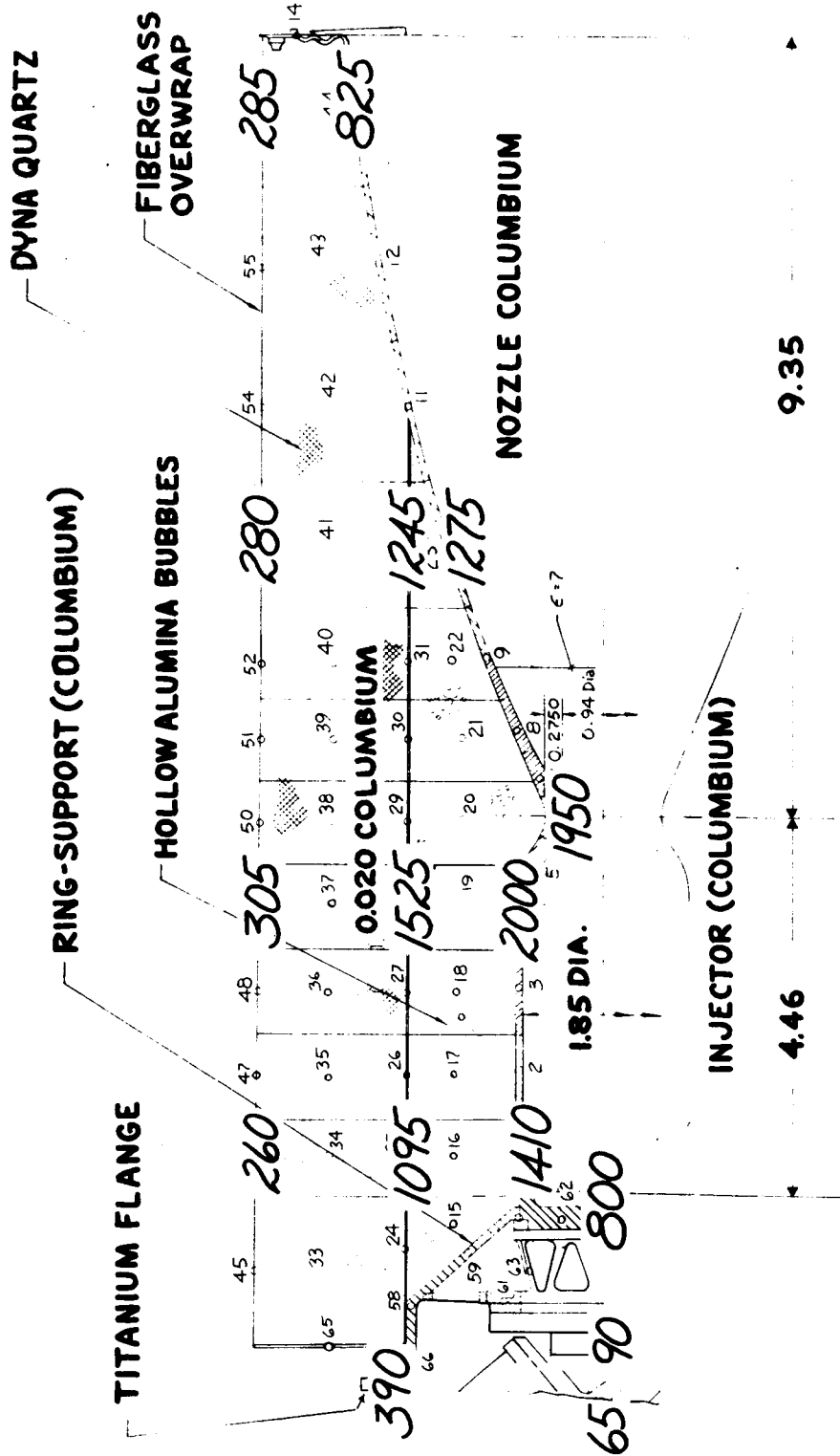


Figure V-172. Thermal Analysis Model - Prototype



STEADY STATE DUTY CYCLE = 1.0

Figure V-173. Thermal Analysis Model - Prototype



STEADY STATE DUTY CYCLE = 0.1

Figure V-174. Thermal Analysis Model - Prototype



Figure V-175. Thermal Analysis Model - Prototype

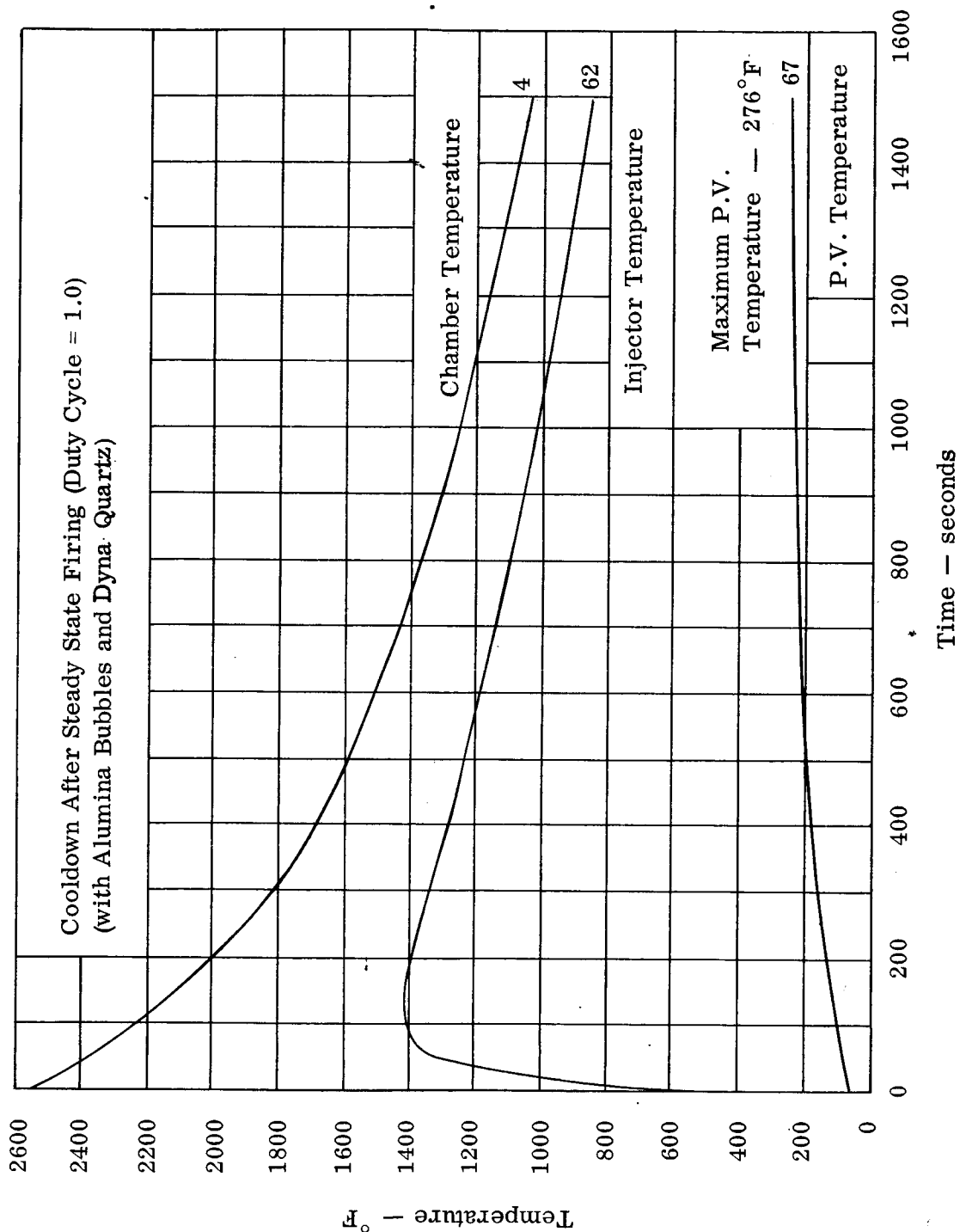


Figure V-176. Chamber, Injector, and Propellant Valve Temperature Time History

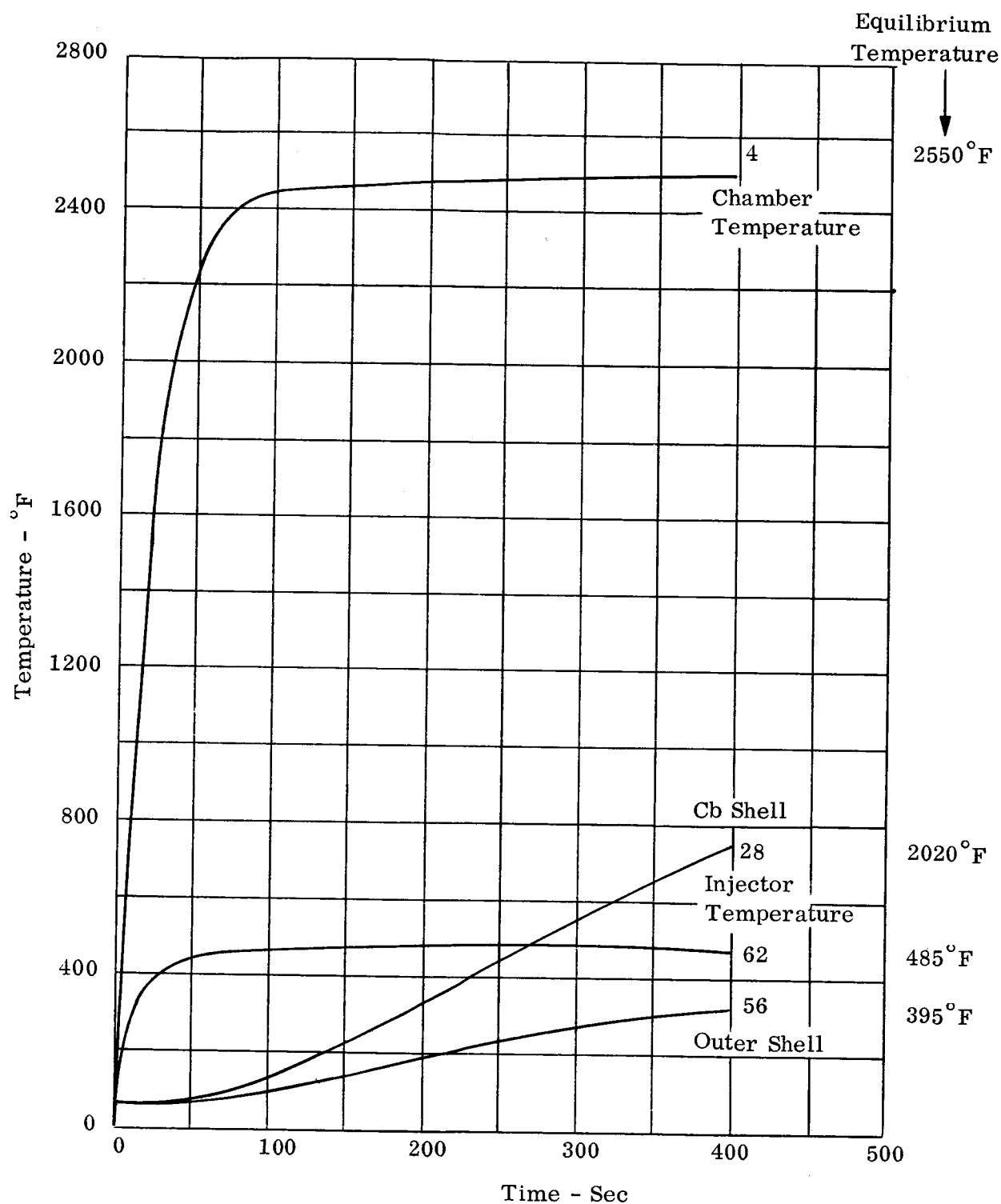


Figure V-177. Temperature Time History (On Time/Total Time = 1.0)

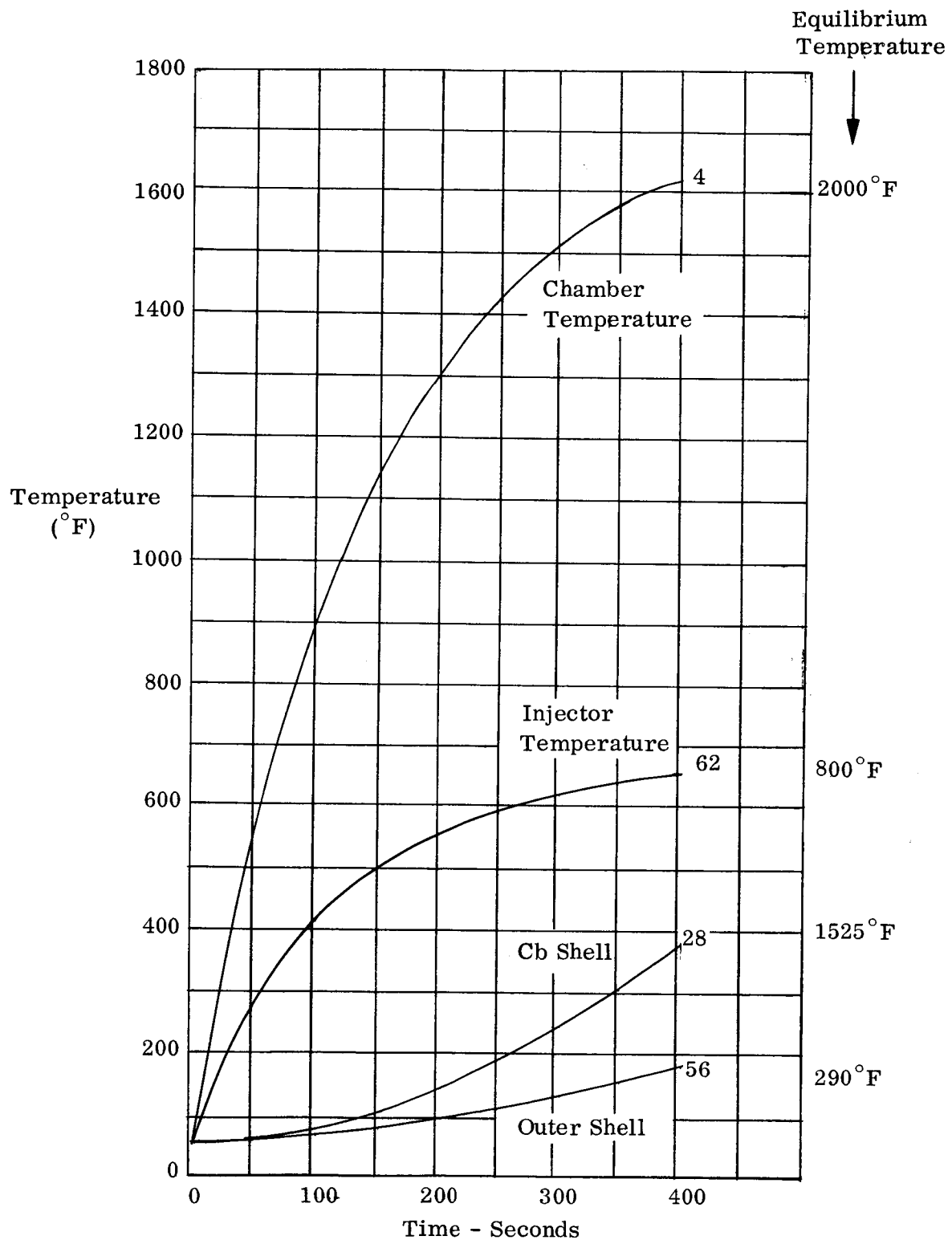


Figure V-178. Temperature Time History (on Time/Total Time = 0.1)

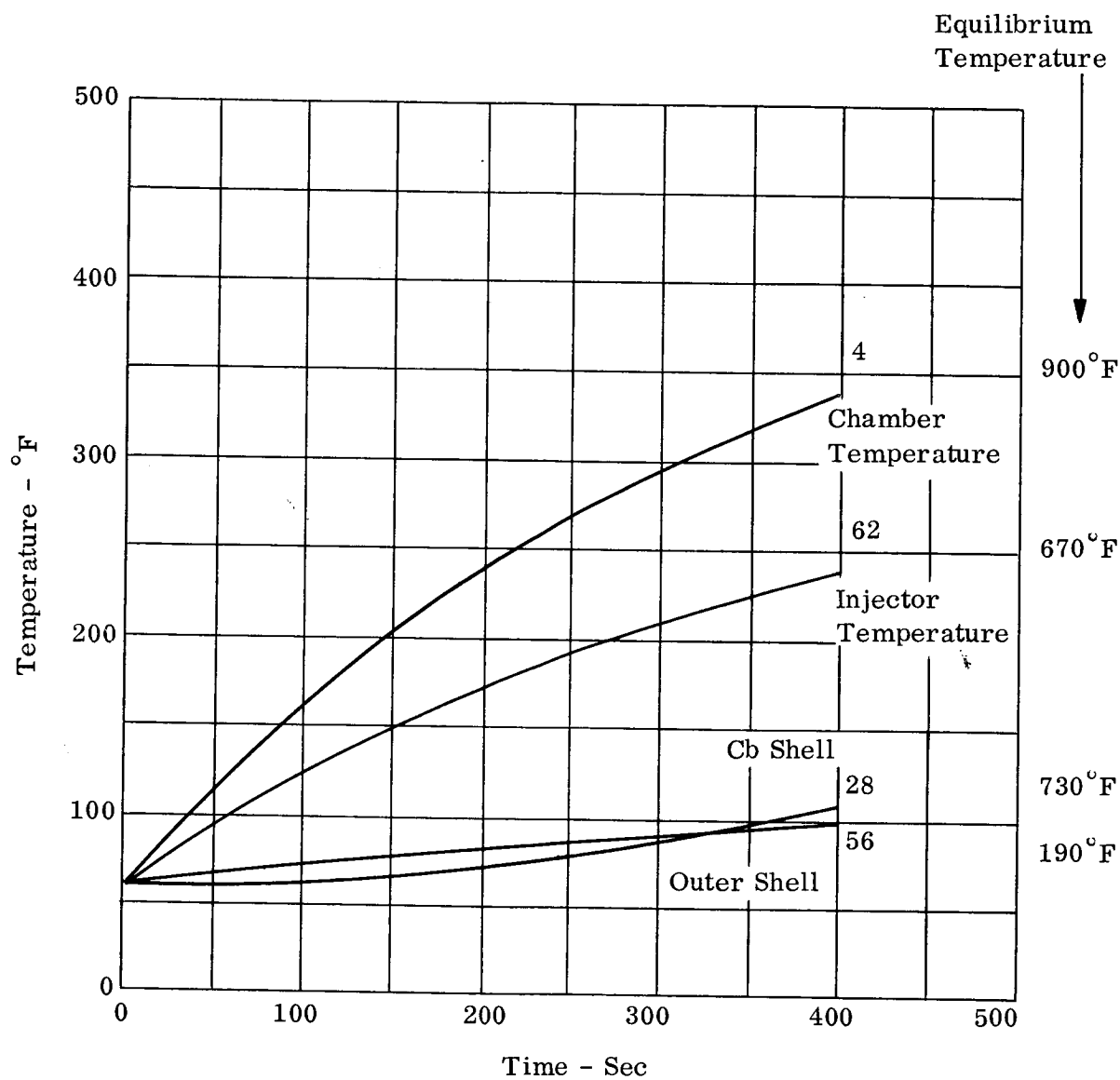


Figure V-179. Temperature Time History (On Time/Total Time = 0.01)

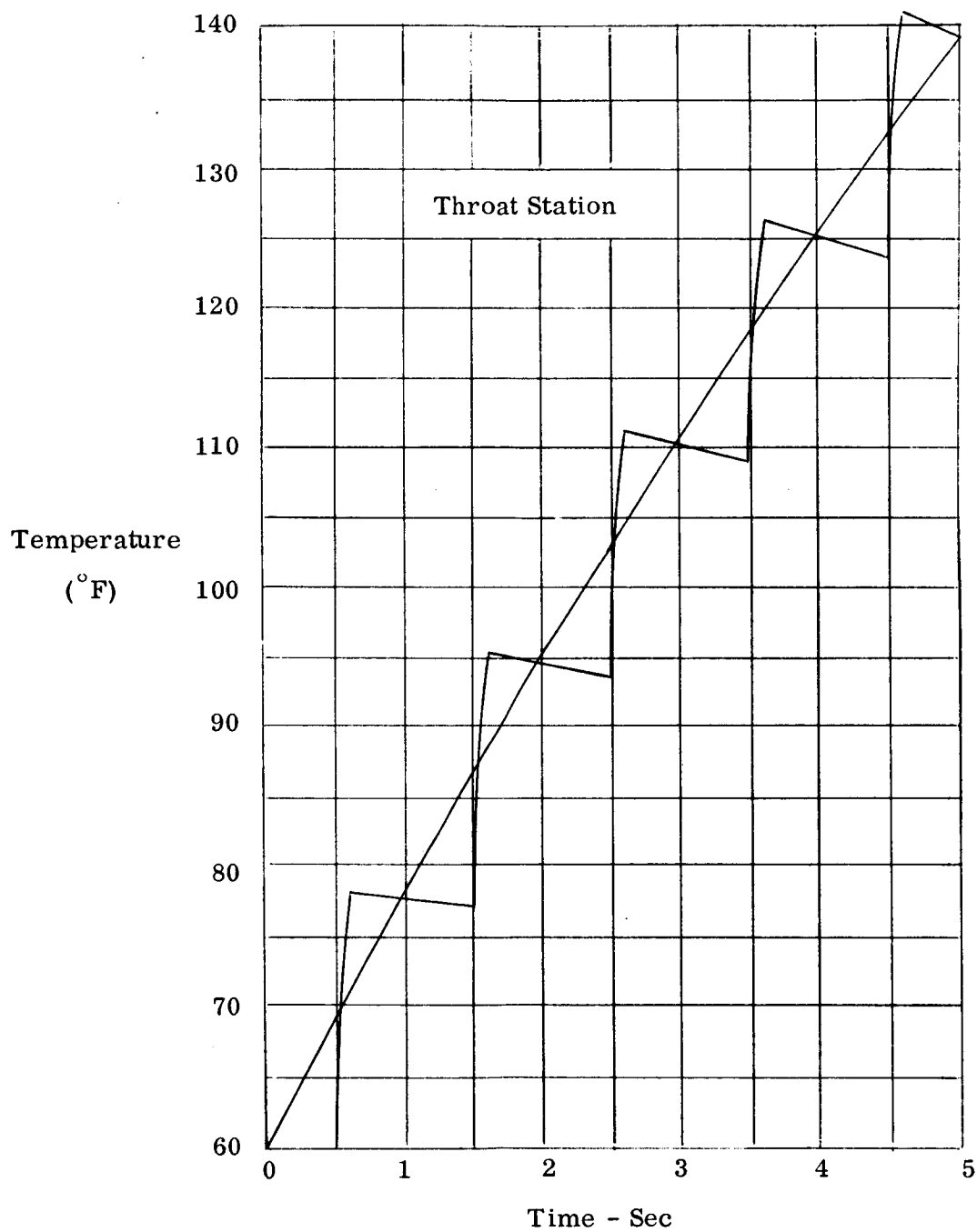
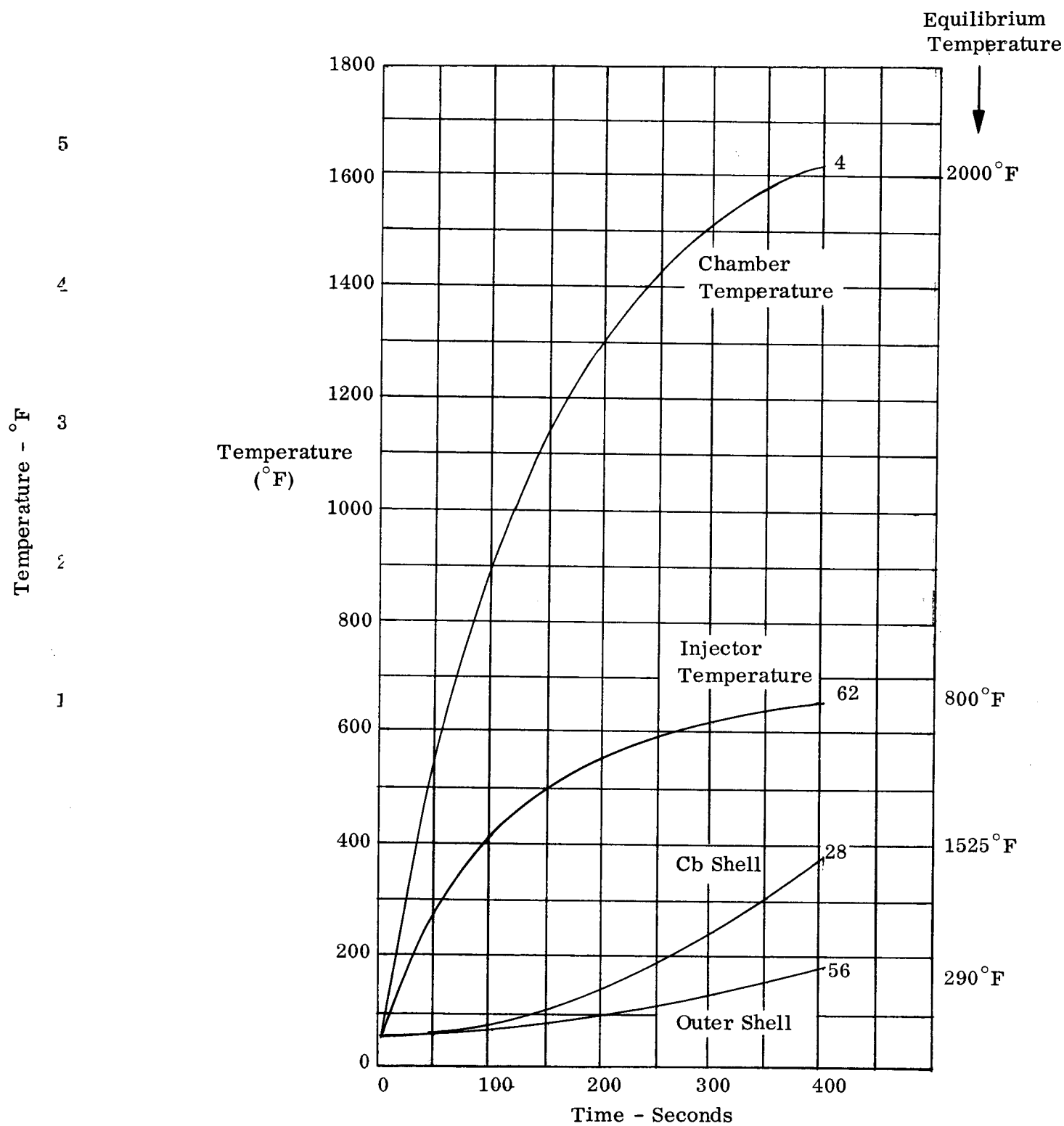


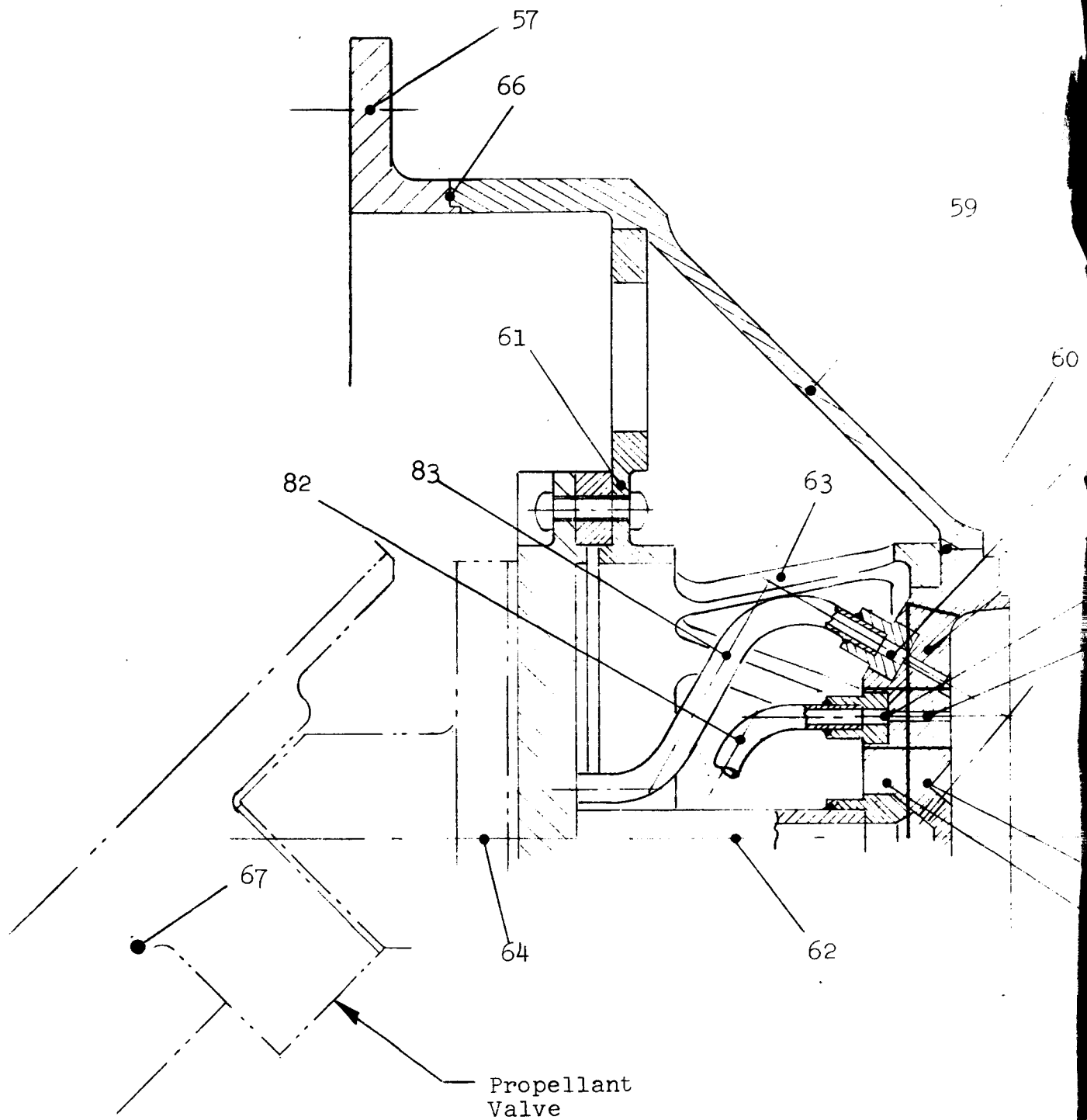
Figure V-180. Temperature Time History (On Time/Total Time = 0.1)





Rep

Figure V-178. Temperature Time History (on Time/Total Time = 0.1)
Report No. 8374-933004



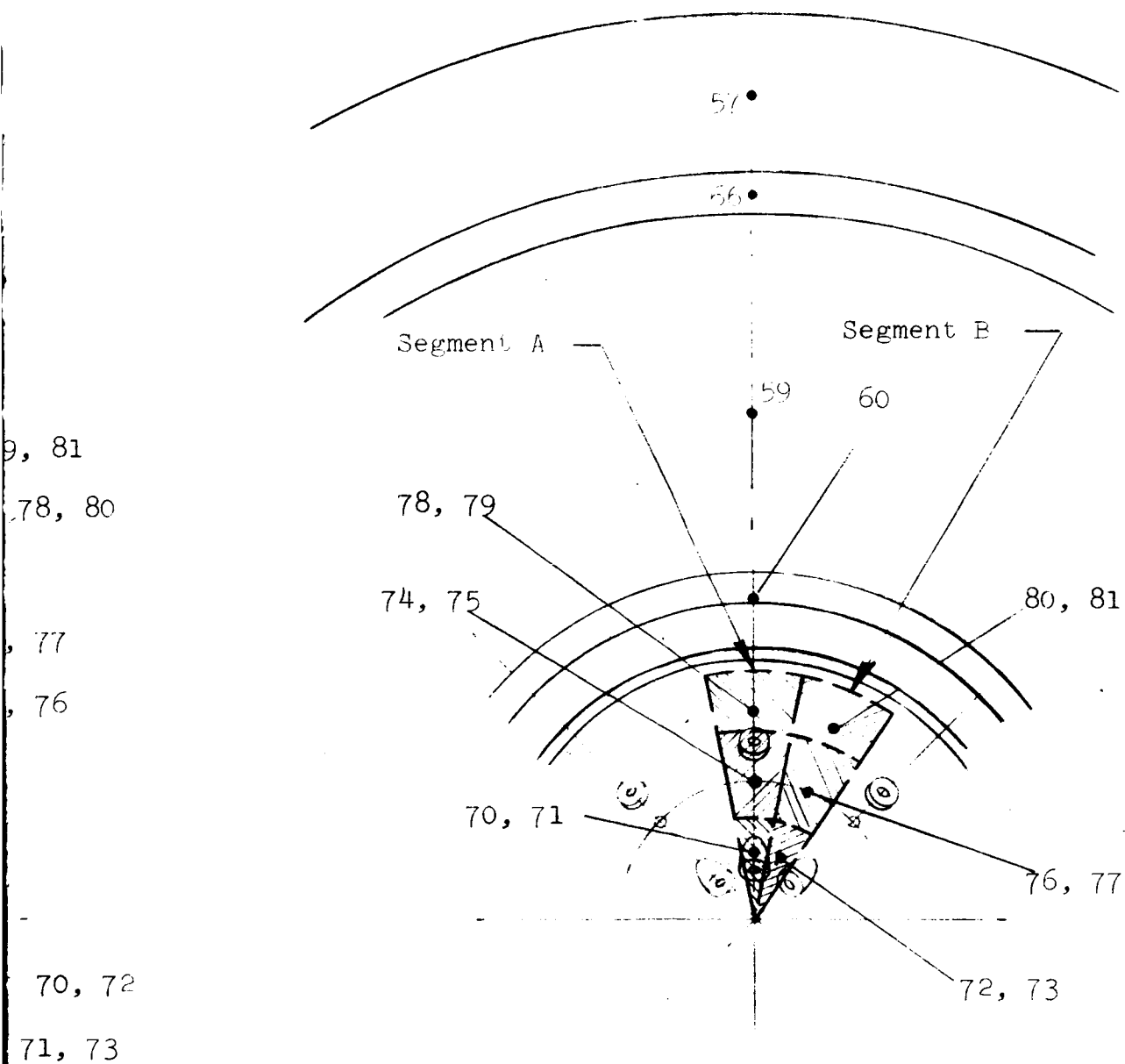


Figure V-181. Thermal Analysis Model - Prototype Injector

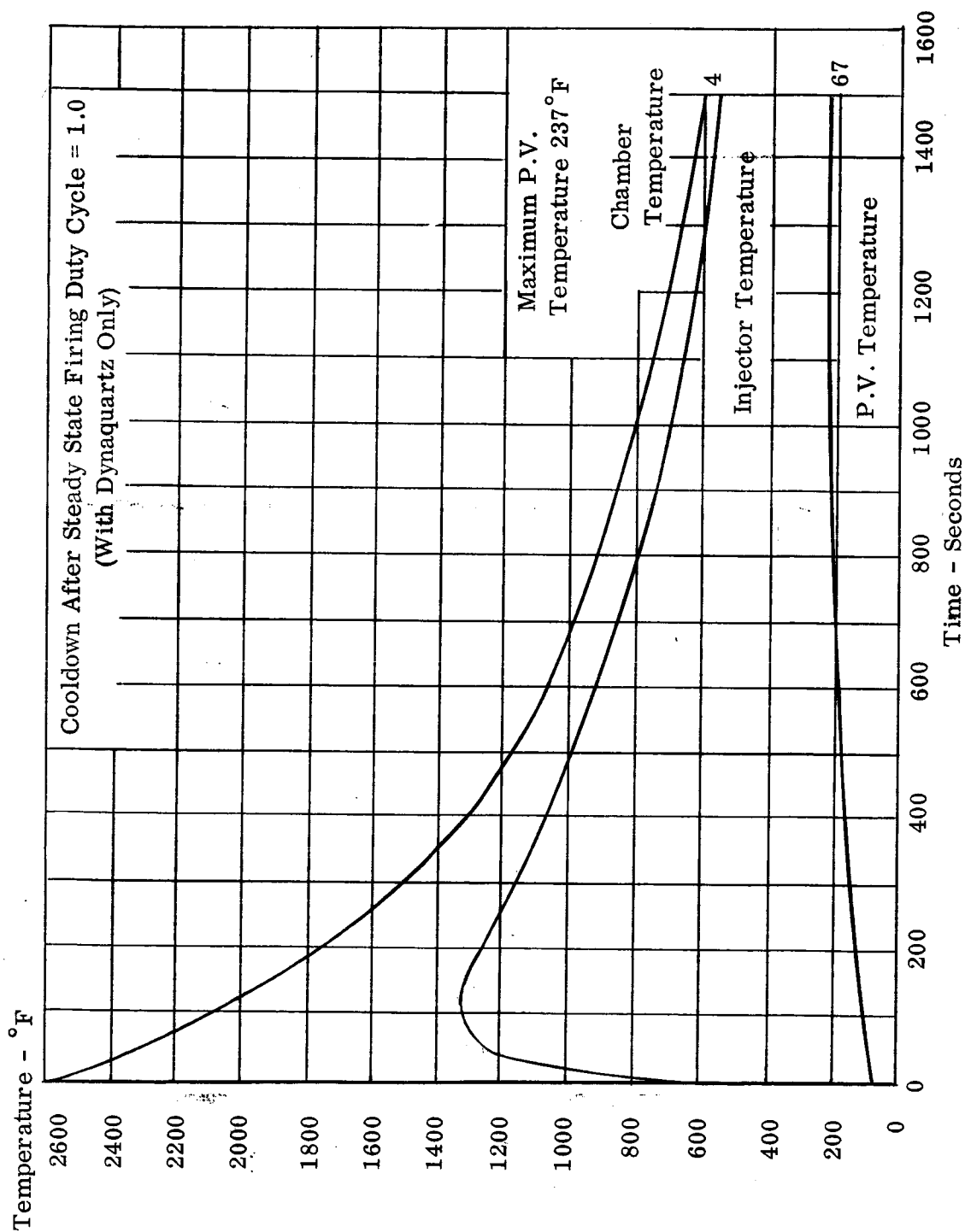


Figure V-182. Chamber, Injector and Propellant Valve Temperature Time History

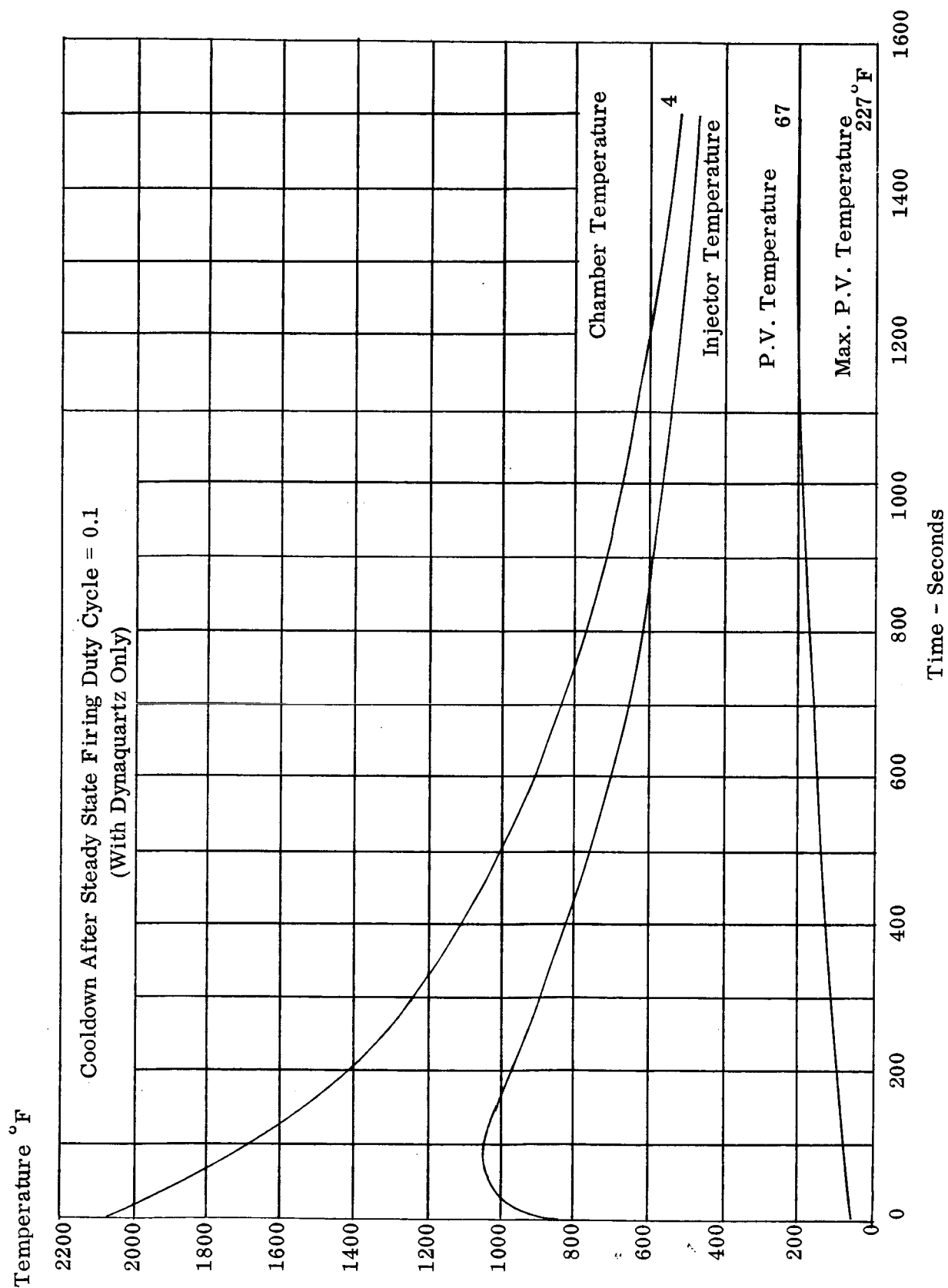


Figure V-183. Chamber, Injector and Propellant Valve Temperature Time History

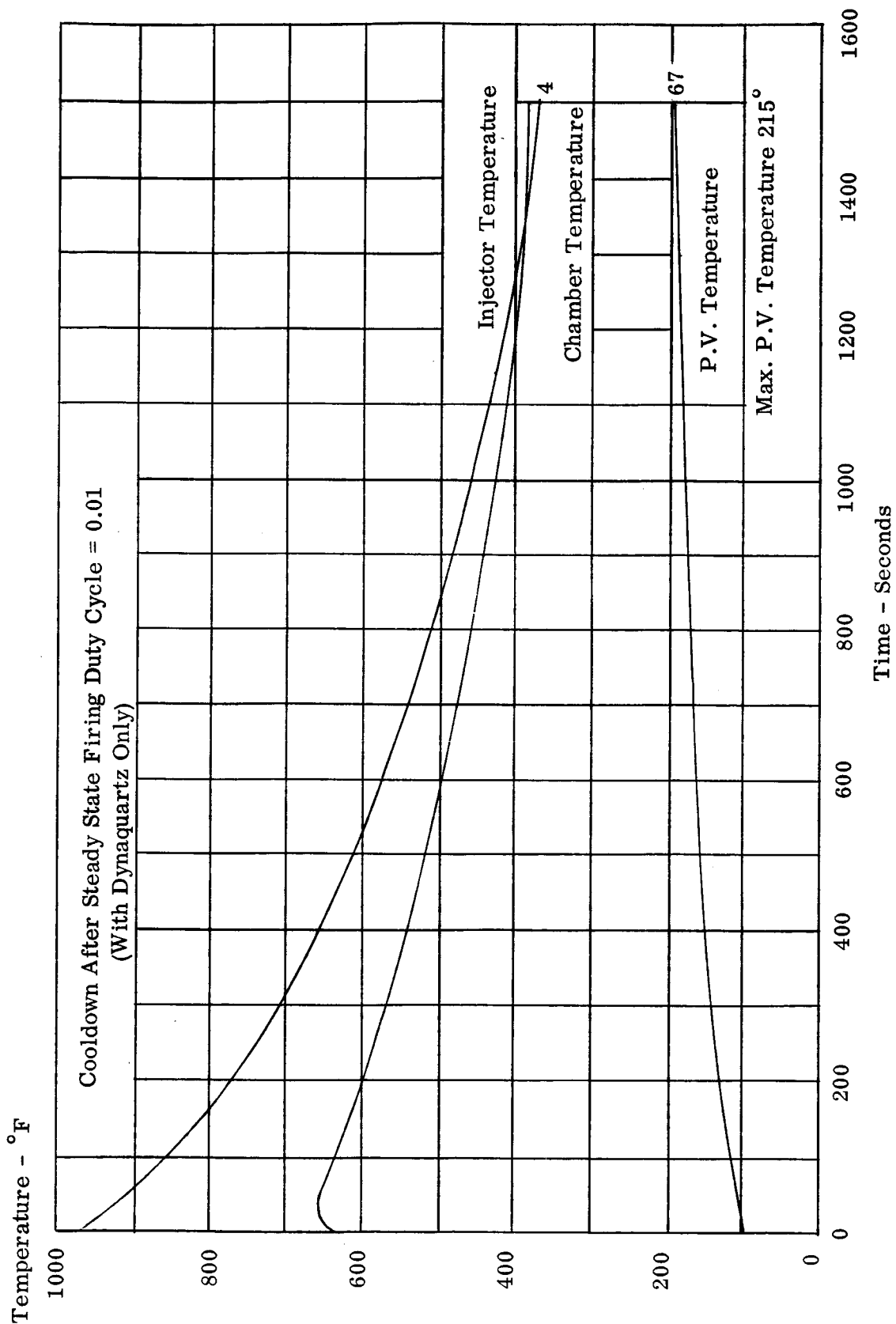


Figure V-184. Chamber, Injector and Propellant Valve Temperature Time History

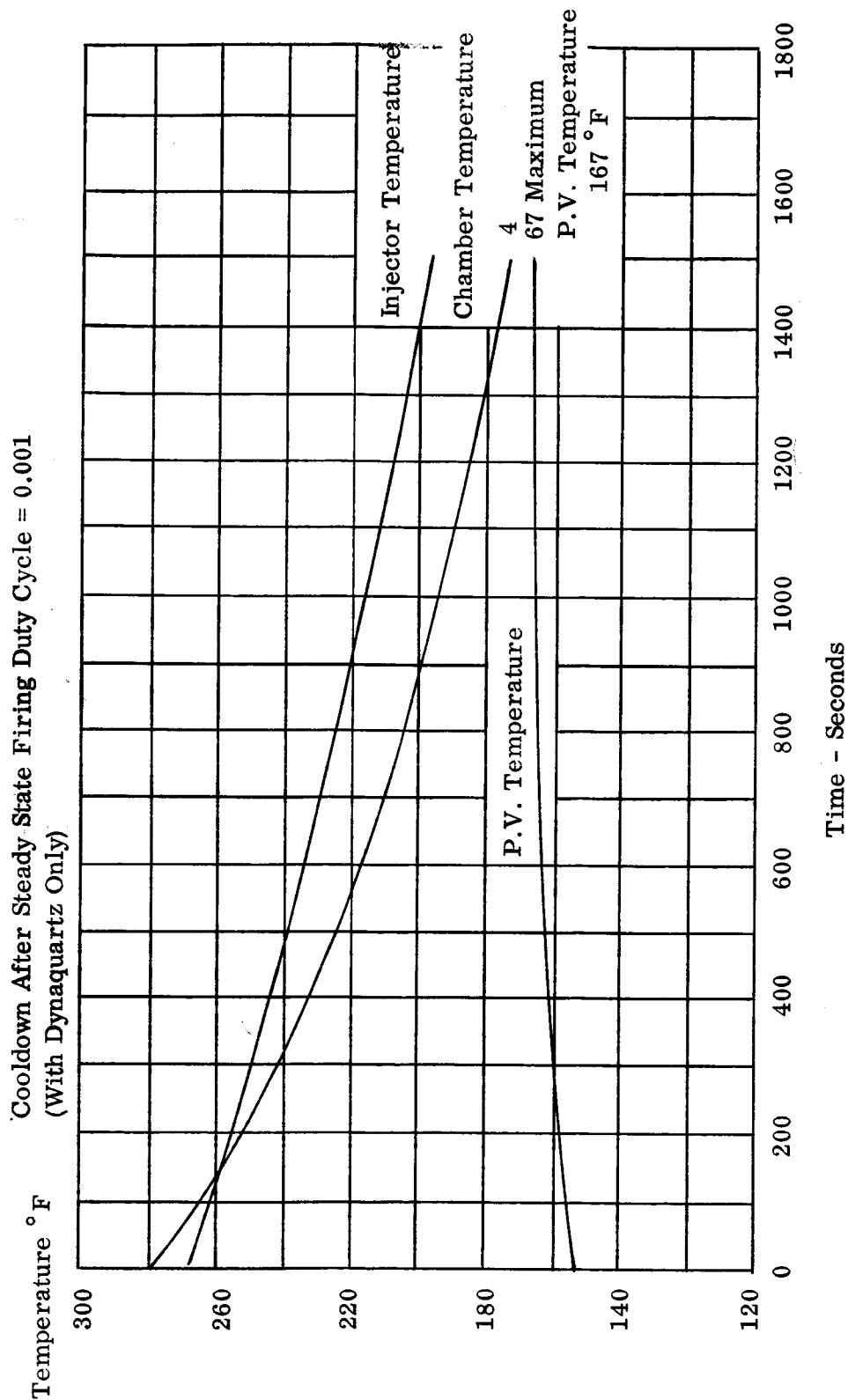


Figure V-185. Chamber, Injector and Propellant Valve Temperature Time History

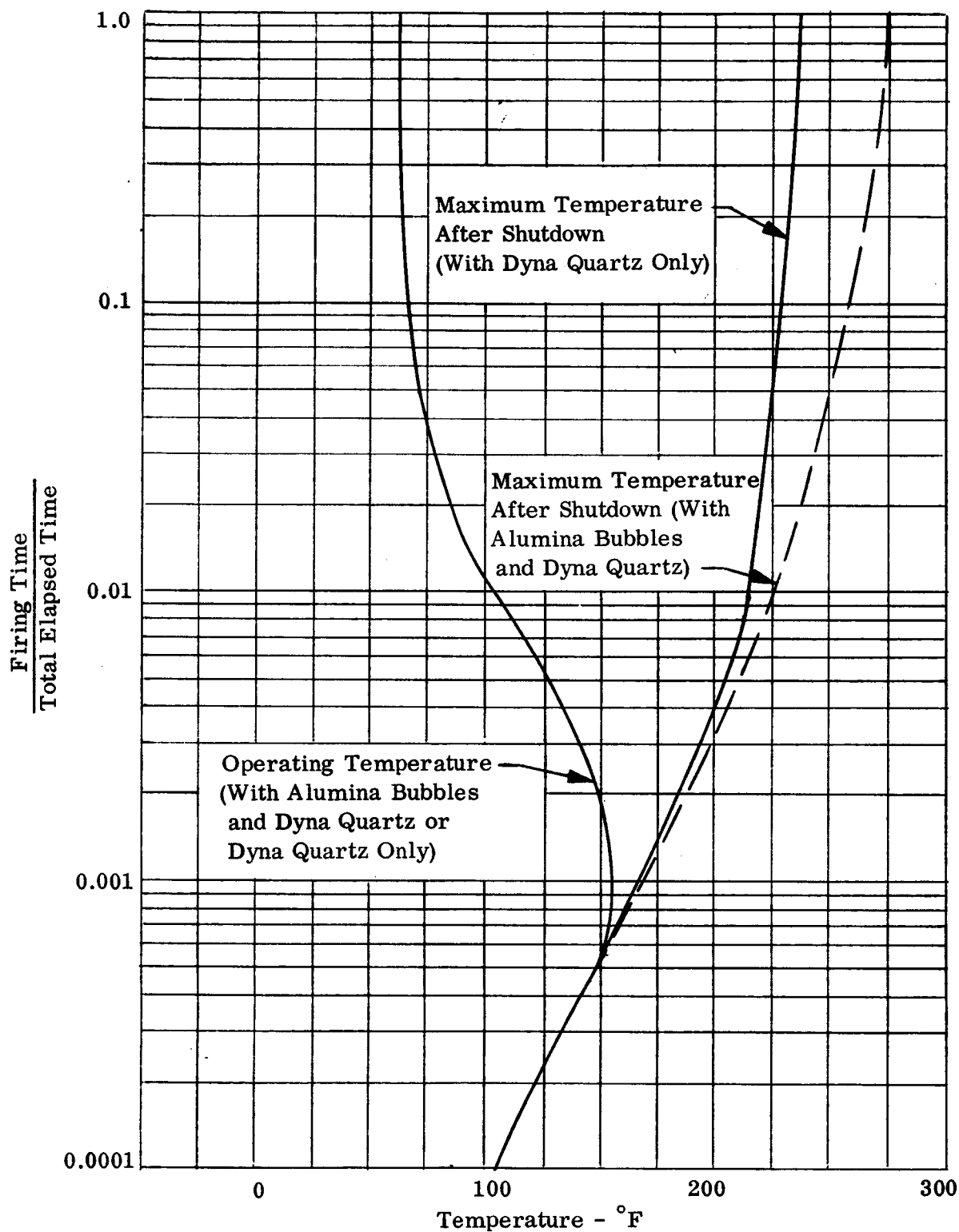


Figure V-186. Injector Temperature versus Duty Cycle

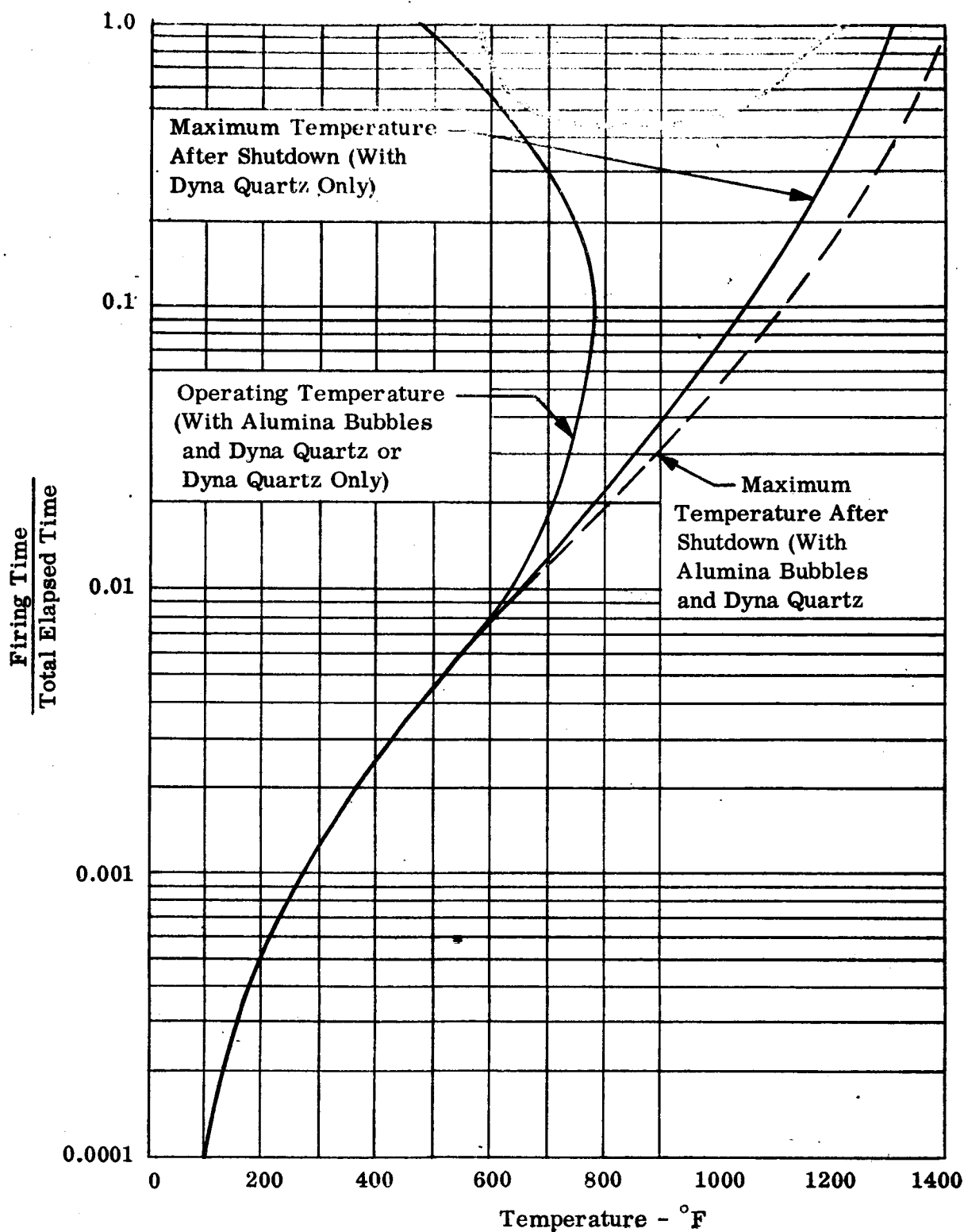


Figure V-187. Propellant Valve Temperature versus Duty Cycle

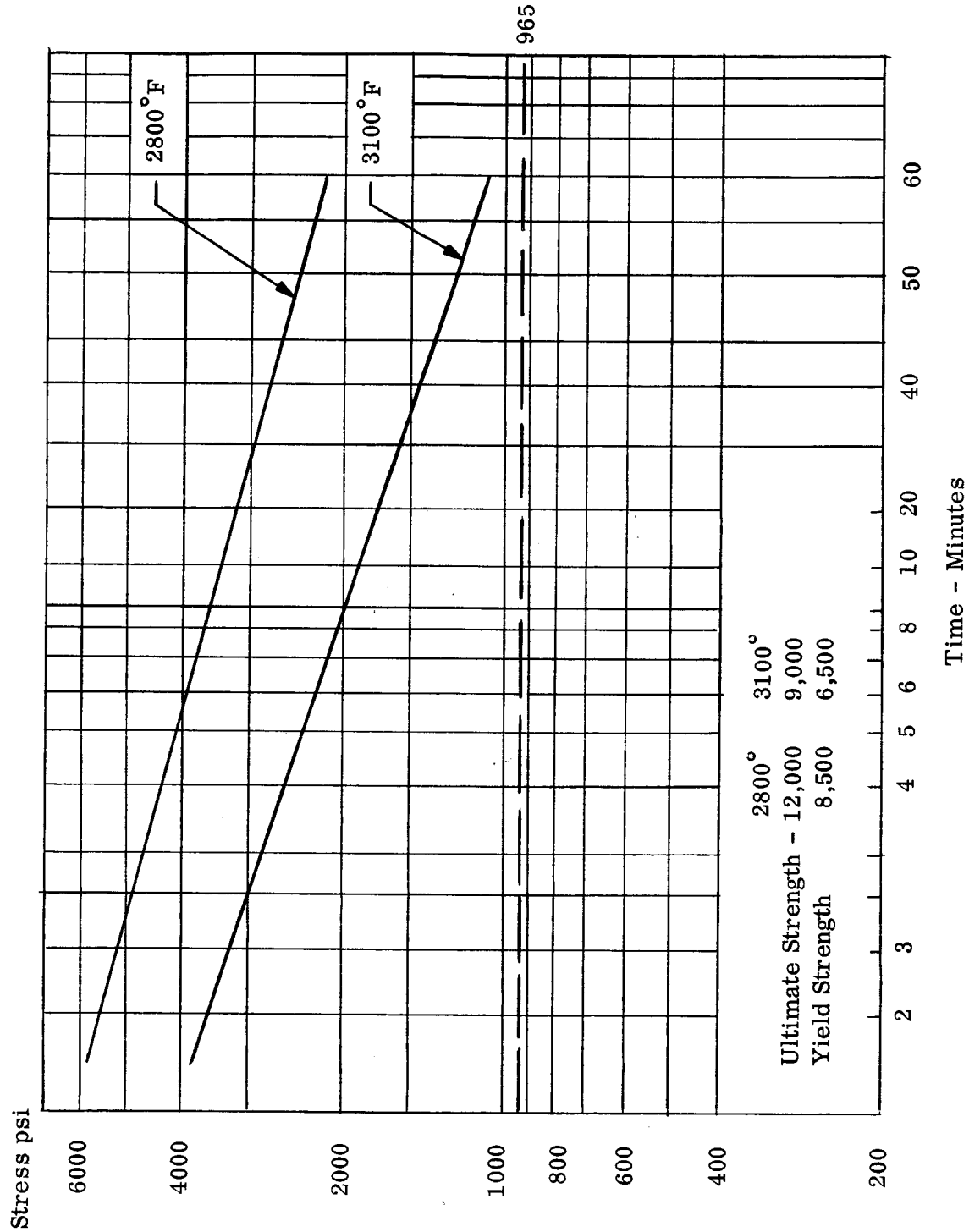


Figure V-188. Time for 0.2% Creep at Various Stress Levels - SCb 291

I. THROTTLING INVESTIGATIONS

1. Throttleable Maneuvering Engine

A company-sponsored development of a throttleable maneuvering engine was initiated during 1964 in support of a Lunar Flying Vehicle (LFV) Study contract with NASA-MSFC.

The objective of this development was a simple and reliable 100-pound maneuvering engine capable of efficient throttling and pulsing operation on N_2O_4 -0.5 N_2H_4 /0.5 UDMH propellants. Optimization studies indicated that radiation cooled engines designed for 80 to 100 psia chamber pressure and the minimum possible propellant feed pressures result in minimum weight/maximum performance maneuvering propulsion systems. Therefore, this engine was designed for 200 psia feed pressures and incorporated a columbium radiation cooled thrust chamber operating at 85 psia chamber pressure to provide a close practical approach to the optimum. The engine incorporates a cavitating venturi type bipropellant throttling valve to provide a thrust range of 5 to 1. A close coupled, fast response bipropellant valve, in combination with the low L^* combustion chamber, provides efficient pulse operation capability. The nozzle expansion ratio was 40 to 1. The engine design is shown in Figure V-189.

This development was based on an existing fixed thrust 100-pound engine (Model 8290) designed for pulsing operation to NASA specifications. The close coupled bipropellant valve, six triplet injector, and columbium radiation cooled thrust chamber design was retained intact. Only injector modification was required to accommodate the design mixture ratio of 1.6 and to provide increased injection pressure drop for efficient and stable throttling. The incorporation of a cavitating venturi type bipropellant throttling valve in tandem with the bipropellant valve completed the maneuvering engine assembly.

One engine was built and tested on a breadboard configuration. The 14-run test program totaled 403 sec running time on the engine. This included 341 sec operation at approximately 125,000 ft altitude during a 3-run test series in an altitude cell. The engine throttling performance obtained at this altitude over a 5 to 1 throttling range is shown in Figure V-190. This figure gives the measured specific impulse, corrected to infinite altitude, versus thrust level. The chamber pressure range corresponding to the thrust range shown in this figure was 16 to 85 psia. Over the total thrust range, the throttle valve demonstrated a mixture ratio control tolerance of $\pm 3\%$ about the mean.

Engine operation throughout these tests, in stepped and continuous throttling, was smooth, stable, and reproducible.

This prototype engine demonstrated the primary design objective of stable and controllable throttling without resort to gas assist or variable injector geometry. Predicted performance was obtained from full thrust to $2/3$ thrust. However, performance in the low thrust range fell short of the design goal of 275 sec specific impulse at $1/3$ thrust. This was attributed in part to the low 15 L* combustion chamber retained to maximize response in pulse mode operation, and partly to a fabrication variation that resulted in injection pressure drops lower than design values. It was concluded that this performance goal could be attained with further development in these areas.

2. Fox Throttle Valve

The throttle valve employed on this engine is a bipropellant variable area cavitating venturi valve procured from the Fox Valve Development Company. A drawing and schematic of the valve are presented in Figure V-191. A photograph of this valve is shown in Figure V-192.

The valve design employs two cavitating venturies in which a needle-shaped pintle is inserted into the venturi throat. Axial movement of the pintle in the throat varies the throat area, producing the means of flow modulation. The fuel and oxidizer venturi pintles are connected to a common yoke and are selectively matched and contoured to provide a 1.6 mixture ratio. The pintles, through the yoke, are positioned by a double acting hydraulic piston utilizing fuel as the control fluid. This piston is piloted by two high response, normally closed solenoid valves. When the actuation circuit is primed with fuel, the valve is liquid locked in any preset position. Energization of the "open" solenoid allows fuel pressure to enter the bottom side of the actuator piston. The actuator then drives the valve open, due to the differential area. Energization of the "close" solenoid vents the fuel pressure in the bottom side of the actuator and allows the source pressure to drive the valve closed. Thus, the throttle position is dependent upon the number and/or duration of energization pulses, and also upon the restriction of the actuator feed lines. A needle-type shutoff valve is incorporated into the actuator feed line for opening the valve. Adjustment of this valve effects the orificing of the actuator feed line and thereby provides a means of setting pulse resolution or response regulation for the



valve opening cycle. A similar type valve may be attached to the actuator vent port for controlling the valve closing cycle.

3. Test Program

The objective of the test program was to experimentally evaluate the throttling characteristics and performance of the Model 8414 maneuvering engine. The pulse operation capability of this design was evaluated on the fixed thrust predecessor Model 8290 engine.

The test program was carried out in the low thrust altitude cell (2ES) at Bell Test Center. The 125,000 ft altitude capability of this test cell was required to maintain full flow (no separation) of the 40/1 nozzle to obtain full flow thrust measurements over the design chamber pressure range of 16 to 85 psia. Figure V-193 shows the test engine installed in the altitude chambers.

Thrust was measured by a load cell to a 3σ error of $\pm 1\%$ of rated. Chamber pressure was measured by double transducers to a 3σ error of $\pm 0.5\%$ of rated. Fuel and oxidizer flow rates were measured by single flowmeters each to a 3σ error of $\pm 0.7\%$ of rated flows. The thrust load cell calibrations before the test series and with propellants after the test series were in close agreement.

The testing began with a 5 run series at sea level totaling 32 sec run time to establish calibration data and trimming requirements. The injector pressure drops (fuel and oxidizer) were lower than expected. Orifices of 0.088 and 0.089 inches diameter, respectively, were then installed in the fuel and oxidizer sides of the injector adapter inlet to provide a pressure drop equivalent to the Moog valve (replaced here by a 5 psi ΔP bipropellant valve) and to balance the total injection ΔP on the fuel and oxidizer sides of the engine.

Testing for performance data then proceeded with a 6 run sea level test series, of 5 sec each, to evaluate performance and stability through a range of fixed throttle settings down to the lowest chamber pressure (e.g., 36 psia) for which sonic flows could be expected in an engine exhausting to ambient pressure.

The program was concluded with 3 runs, at approximately 125,000 ft altitude, of 43, 63, and 235 seconds duration, respectively. These tests included stepped and continuous throttling, starts at 100% thrust and 38% thrust, and shutdown at 100% and 20% thrust to span the spectrum of throttled operating conditions.

4. Test Results

The results of the fixed throttle setting tests at sea level are shown in Figure V-194. Throttled operation was smooth and stable with the exception of the 5th run (2ES 1662) which toned audibly at a P_c of 44 psia. Since this did not occur at chamber pressures above or below this value or at this value in the subsequent altitude tests, it was attributed to unstable flow separation in the nozzle.

The first altitude throttling test results are shown in Figure V-195. The engine was continuously throttled down to 28 lb thrust, stabilized, and then throttled at a faster rate back to 98 lb thrust. Comparison of P_c and thrust with throttle valve position shows the full open "deadband" of this prototype valve between 0.46 and 0.56 inch open. A production valve would be "stopped" at 0.46 inch to eliminate this deadband. The low ambient temperature prevailing at the time of the test resulted in a cold test of the engine with a fuel temperature 8° F above the freezing point. Specific impulse was within the range expected of the design and throttling was smooth and stable.

The results of the second altitude test are shown in Figure V-196. The engine was started at 38 lb thrust, step throttled up to 92 lb, and continuously throttled back to 20 lb thrust. Throttling was smooth and stable in both the stepped and continuous modes.

Propellants were conditioned to near room temperatures for the third altitude test. The objective of this test was to obtain performance and durability data during stabilized operation at all points in the design thrust spectrum. The results of this 235 sec altitude test are given in Figure V-197. The engine was started at 97 lb thrust, stabilized for 63 sec to allow the radiation cooled thrust chamber to come to thermal equilibrium, then step throttled down to 18 lb thrust and back to 98 lb thrust where it was operated for an additional 79 sec to demonstrate durability. Total test time on a single run was limited by steam capacity for the steam ejectors of the altitude cell. Throttling was smooth and stable throughout the test.



5. Throttling Demonstration - TU-6A and TU-3B Injector

a. TU-6A Injector

The unbalanced triplet injector TU-6A was fire-tested in a radiation-cooled columbium chamber at sea level conditions with propellant combinations of N_2O_4 /50% N_2H_4 - 50% UDMH (50-50 blend) and N_2O_4 /MMH.

This test series was conducted with Company-sponsored funding to investigate the throttling capability of the unbalanced injector using both propellant combinations, and to determine the injector performance and chamber surface temperatures using 50-50 blend as the fuel. The Fox variable area, cavitating venturi bipropellant valve was used for the entire series.

This series consisted of a total of seven tests, four tests being conducted with the propellant combination N_2O_4 /50-50 blend, and three tests with the propellant combination N_2O_4 /MMH. The test series using 50-50 blend as the fuel consisted of one 5-second and one 10-second duration check run at rated chamber pressure of 80 psia, one 48-second duration test at rated chamber pressure with chamber temperatures stabilized, and one 25-second duration test where the injector was throttled from 80 psia chamber pressure to 50.9 psia chamber pressure.

The steady state performance of this injector, using 50-50 blend as the fuel, was approximately 3% higher at rated mixture of 1.6 than that obtained using MMH as the fuel. The comparison of c^* is approximately 5585 ft/sec using 50-50 blend, and approximately 5400 ft/sec using MMH. The chamber wall temperatures were approximately 400°F higher with the 50-50 blend than that obtained with MMH.

Past experience using balanced triplet injectors with 50-50 blend has not revealed this considerable temperature increase since the difference in theoretical gas temperatures is relatively small for 50-50 blend and MMH at nominal mixture ratios (1.6 to 2.0). The theoretical combustion temperature for N_2O_4 /MMH propellants at a mixture ratio of 0.8 is 400°F lower than the temperature obtained at the same mixture ratio with N_2O_4 /50-50 blend,

as shown in Figure V-198; therefore, a significant temperature difference can be expected when using an injector with a fuel-rich barrier mixture ratio such as TU-6A. As indicated from the theoretical values presented in Figure V-199, a corresponding increase in performance is also expected.

It is most probable that an iteration can be made to the TU-6A injector design that would result in a similar temperature-performance level with N_2O_4 /50-50 blend as presently experienced with N_2O_4 /MMH. This iteration would consist of additional unbalance of the injector to reduce the barrier mixture ratio, thereby reducing the temperature.

Throughout the throttling range of 80 psia to 50.9 psia, no instability was noticed either audibly or with instrumentation. Close-coupled, high frequency response instrumentation could not be used since chamber pressure was measured at the injector face.

A tabulation of the test data for this series is shown in Table V-34.

The test series using MMH as the fuel consisted of one 5-second and one 15-second duration check run, and one 102-second duration test. The first 60 seconds of this test was at rated chamber pressure of 80 psia; the remaining 42 seconds consisted of throttling to 17.8 psia. Again, performance was stable throughout the throttling range. The performance level and chamber temperature level compared well with that obtained during previous testing of this injector with MMH.

A tabulation of the test data for this series is also shown in Table V-34.

b. TU-3B Injector

The unbalanced triplet injector TU-3B was fire-tested in the radiation-cooled, columbium chamber at sea level conditions with the propellant combinations of N_2O_4 /50% N_2H_4 - 50% UDMH.

This test series consisted of three tests, and was conducted with Company-sponsored funding, to investigate the throttling capability and performance and temperature profile of this lower performing injector using 50-50 blend as the fuel. The Fox throttle valve was also used for this series.

TABLE V-34
 INJECTOR TU-6A AND TU-3B PERFORMANCE DATA THROTTLING DEMONSTRATION
 Columbium Chamber - Radiation Cooled
 Data corrected for throat expansion and contraction ratio (P_c pickup location)

INJECTOR TU-6A

Run No. (Cell B-2)	Run Duration (sec)	Data P_T (sec)	c^* corr (ft/sec)	P_c (psia)	O/F	Fuel	Nozzle Temperature ($^{\circ}$ F)		
							Min.	Ave.	Max.
2246	5	5.0	-	83.1	1.57	50-50	-	-	-
2247	10	10.0	-	81.3	1.55	50-50	-	-	-
2248	48	10.0	-	79.2	1.63	50-50	-	-	-
		30.0	5583	78.6	1.61	50-50	2661	2710	2748
		48.0	5589	78.5	1.61	50-50	2638	2684	2722
2249	25	9.0	-	79.5	1.60	50-50	-	-	-
		14.0	-	75.3	1.59	50-50	-	-	-
		22.0	5540	64.5	1.59	50-50	2506	2560	2613
		25.0	5495	50.9	1.59	50-50	2526	2573	2629
2250	5	4.0	-	78.0	1.62	MMH	-	-	-
2251	15	15.0	-	79.6	1.67	MMH	-	-	-
2252	102	15.0	-	79.3	1.65	MMH	-	-	-
		30.0	5378	79.1	1.65	MMH	2107	2195	2265
		60.0	5370	79.1	1.65	MMH	2050	2190	2279
		78.0	-	73.2	1.63	MMH	2038	2155	2235
		84.0	-	59.9	1.63	MMH	1965	2060	2115
		96.0	-	38.8	1.63	MMH	1704	1790	1853
		102.0	-	17.8	1.66	MMH	1679	1755	1821

TABLE V-34 (CONT)
INJECTOR TU-3B

Run No. (Cell B-2)	Run Duration (sec)	Data P _T (sec)	c* corr (ft/sec)	P _c (psia)	O/F	Fuel	Nozzle Temperature (°F)		
							Min.	Ave.	Max.
2253	5	5.0		73.4	1.53	50-50	-	-	-
2254	25	10.0		-	-	50-50	-	-	-
		25.0	5105	74.5	1.54	50-50	-	-	-
2255	80	30.0	5099	75.6	1.56	50-50	2123	-	2375
		40.0	5162	75.3	1.56	50-50	2136	-	2377
		50.0	-	64.2	1.55	50-50	2141	-	2350
		60.0	-	47.5	1.56	50-50	2040	-	2237
		70.0	-	34.4	1.57	50-50	1820	-	2014
		80.0	-	21.5	1.58	50-50	1672	-	1785

This test series was comprised of one 5-second, one 25-second duration, and one 80-second duration test. The first 40 seconds of the last test was conducted at approximately 75 psia chamber pressure; the remaining 40 seconds consisted of throttling to 21.5 psia chamber pressure. The steady state performance using 50-50 blend as the fuel was approximately 2% higher than with MMH (c^* of approximately 5100 ft/sec versus 4975 ft/sec, respectively). No instability was noted throughout the throttling range.

A summary of these test data is given also in Table V-34.

6. Performance Evaluation (With Other Propellants) - TU-6A Injector

Additional testing was also conducted on the TU-6A injector with Company-sponsored funding, using N_2O_4 /50-50 blend, and N_2O_4 /MMH as the propellant combinations to investigate the performance and temperature level at a mixture ratio of 2.0.

This series consisted of two tests with 50-50 blend, and three tests with MMH. A summary of these test data is given in Table V-35.

A comparison of the performance of TU-6A and TU-3B injectors for the various propellant combinations is given in Figure V-200.

7. Active Cooling Tests (With Other Propellants)

To provide early experimental inputs into the thermal analysis of engine concepts, total heat rejection rates of the existing 100-lb columbium thrust chamber were determined in fire tests. The entire thrust chamber, including the 40:1 area ratio nozzle was enclosed in a water cooled, copper coil shroud in altitude facility 2ES. The thrust chamber and test installation are shown in Figures V-201, V-202, and V-203. The chamber was fired for a duration of 90 seconds, where stabilization of surface temperature and water outlet temperature was obtained.

Two tests were run with propellants N_2O_4 - 50% N_2H_4 + 50% UDMH at mixture ratios 1.56 and 1.85. To aid in the propellant selection and to test the performance of this chamber with another propellant combination, one firing was conducted with MON-UDMH at a mixture ratio 1.77. Results of these tests are tabulated below.

TABLE V-35
TU-6A INJECTOR PERFORMANCE DATA

Columbium Chamber - Radiation Cooled
Mixture Ratio 2.0

Run No. (Cell B-2)	Run Duration (sec)	Data P _T (sec)	c* corr (ft/sec)	P _c (psia)	O/F	Fuel	Nozzle Temperature (°F)		
							Min.	Ave.	Max.
2256	5	5.0	-	79.7	2.01	50-50	-	-	-
2257	60	15.0	5131	79.4	2.02	50-50	2106	-	2385
		30.0	5136	79.6	2.00	50-50	2529	-	2693
		45.0	*	*	2.00	50-50	2511	-	2681
		60.0	*	*	2.00	50-50	2515	-	2670
2258	4	4.0	-	79.1	2.03	MMH	-	-	-
2259	45	15.0	-	78.2	2.03	MMH	1662	-	1974
		30.0	5062	77.6	2.03	MMH	1936	-	2210
		45.0	5048	77.4	2.03	MMH	1965	-	2275
2260	45	15.0	-	77.4	2.03	MMH	1628	-	1966
		30.0	5050	76.8	2.03	MMH	1887	-	2209
		45.0	5033	76.6	2.03	MMH	1878	-	2220
2261	45	15.0	-	77.3	2.00	MMH	1605	-	1980
		30.0	5081	77.1	2.00	MMH	1838	-	2208
		45.0	5100	77.3	2.00	MMH	1912	-	2236

* Chamber pressure leak - data not reliable.

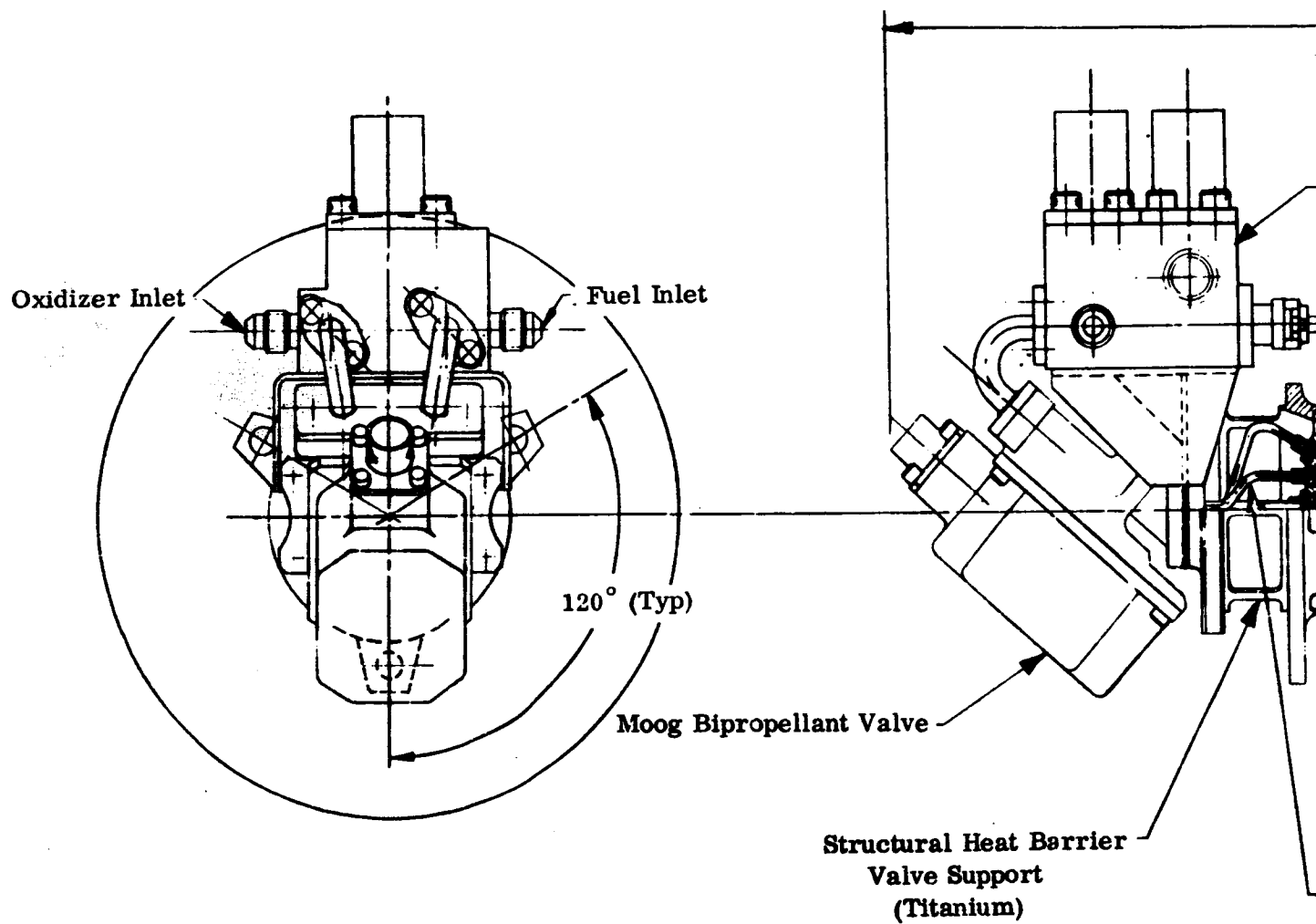
NOTE: Data corrected for throat expansion and contraction ratio (P_c pickup location)

ACTIVE COOLING TESTS

<u>Run No.</u>	<u>Mixture Ratio O/F</u>	<u>P_c (psia)</u>	<u>c* (ft/sec)</u>	<u>Coolant Flow H₂O (lb/sec)</u>	<u>Max Wall Temp Throat (°F)</u>	<u>Total Heat Rejection (BTU/sec)</u>
Prop. N ₂ O ₄ + 50% N ₂ H ₄ - 50% UDMH						
2ES 1647	1.56	83.8	5552	0.188	2650	13.1
2ES 1648	1.85	78.2	5313	0.158	2580	11.2
Prop. MON 10% NO + UDMH						
2ES 1651	1.77	81.4	4485	0.182	1840	3.8

With the propellant combination MON-UDMH a reduction in combustion efficiency from 94% to 80% was observed with corresponding reduction in wall temperatures and total heat rejection. Indications are that the injector design is not optimized for this propellant combination but a check firing with N₂O₄ 50/50 blend was performed later in the program.

The results of these tests are being used to confirm and adjust as necessary the present detailed analysis for steady state operation.



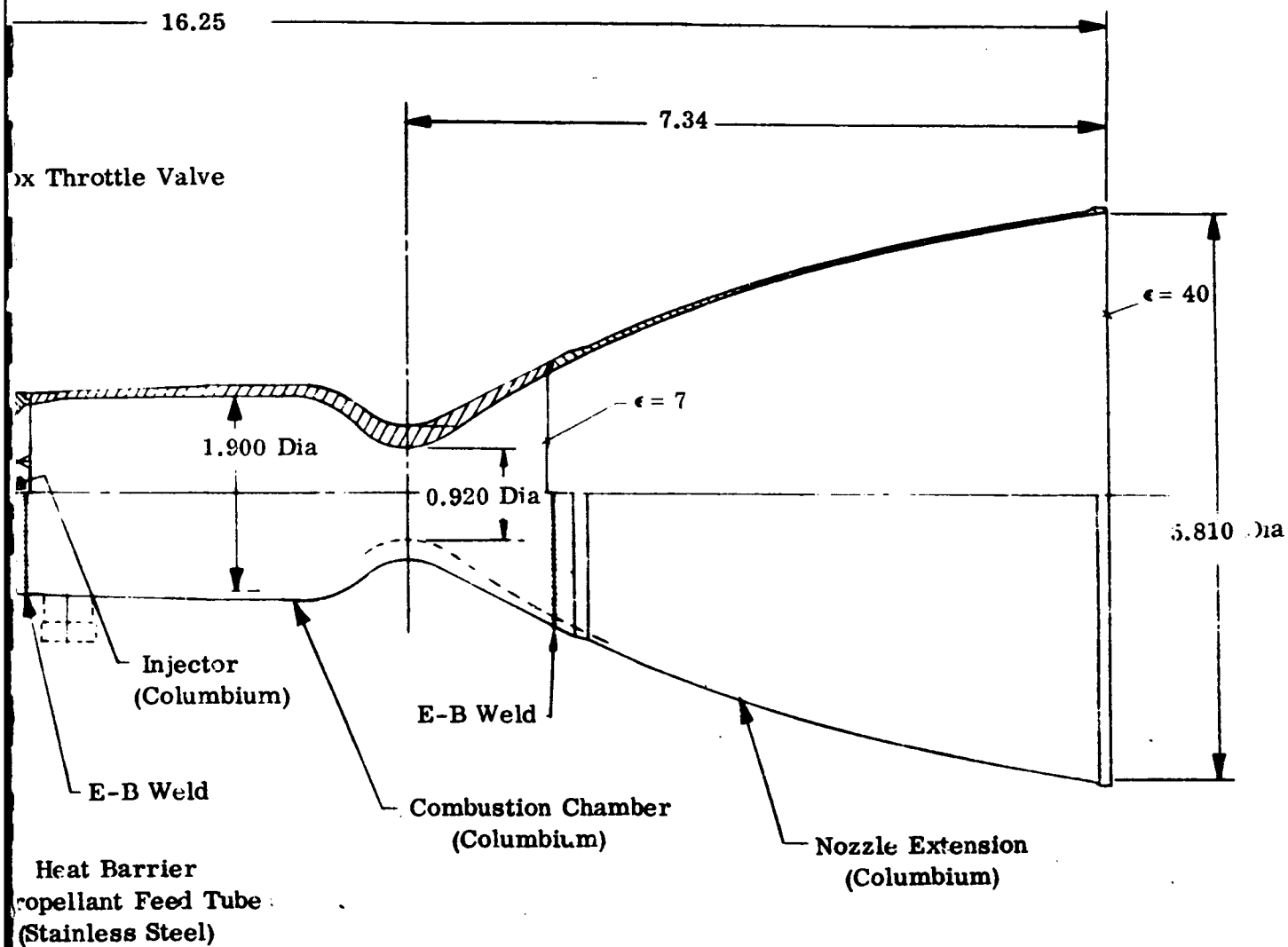


Figure V-189. Prototype Maneuvering Engine

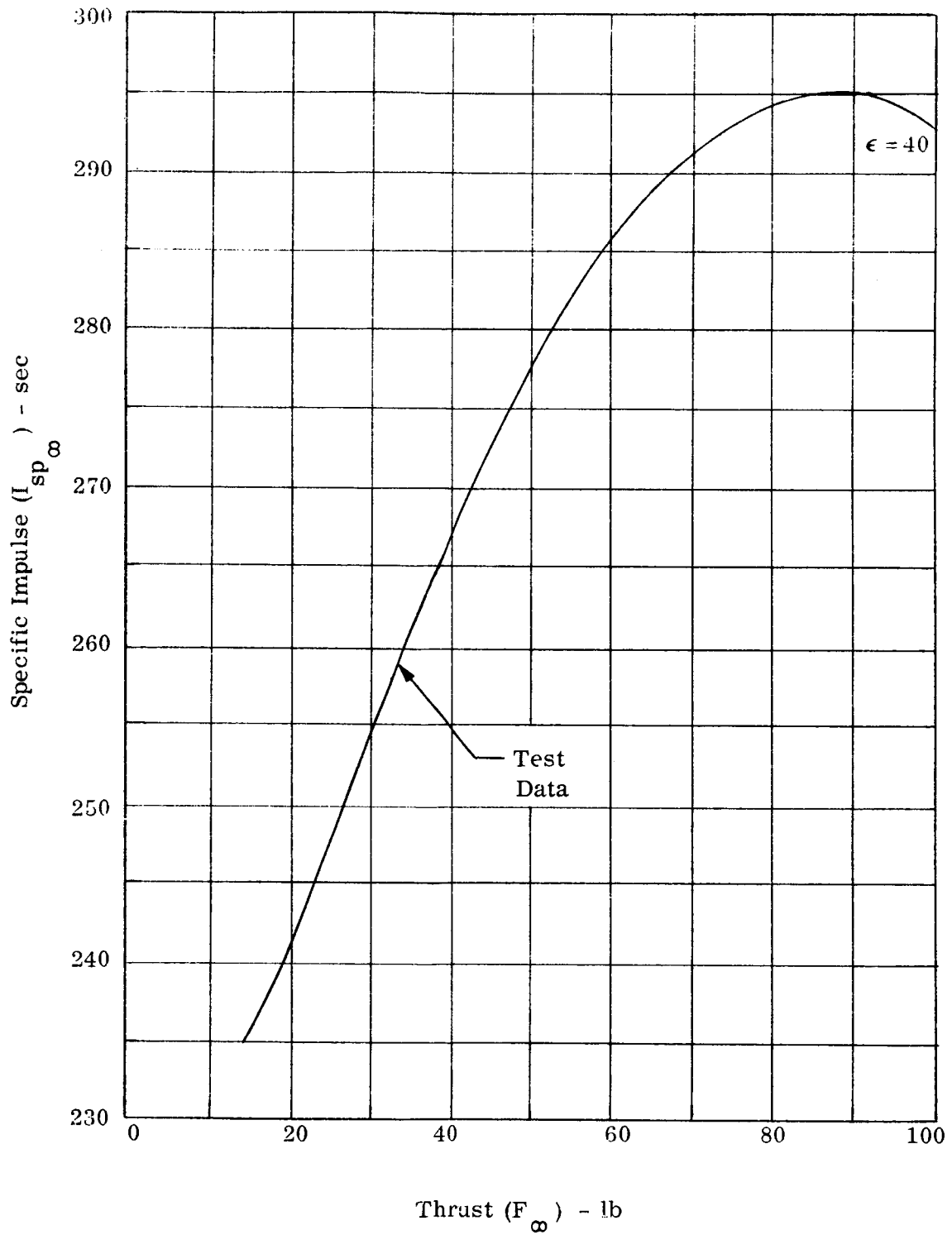
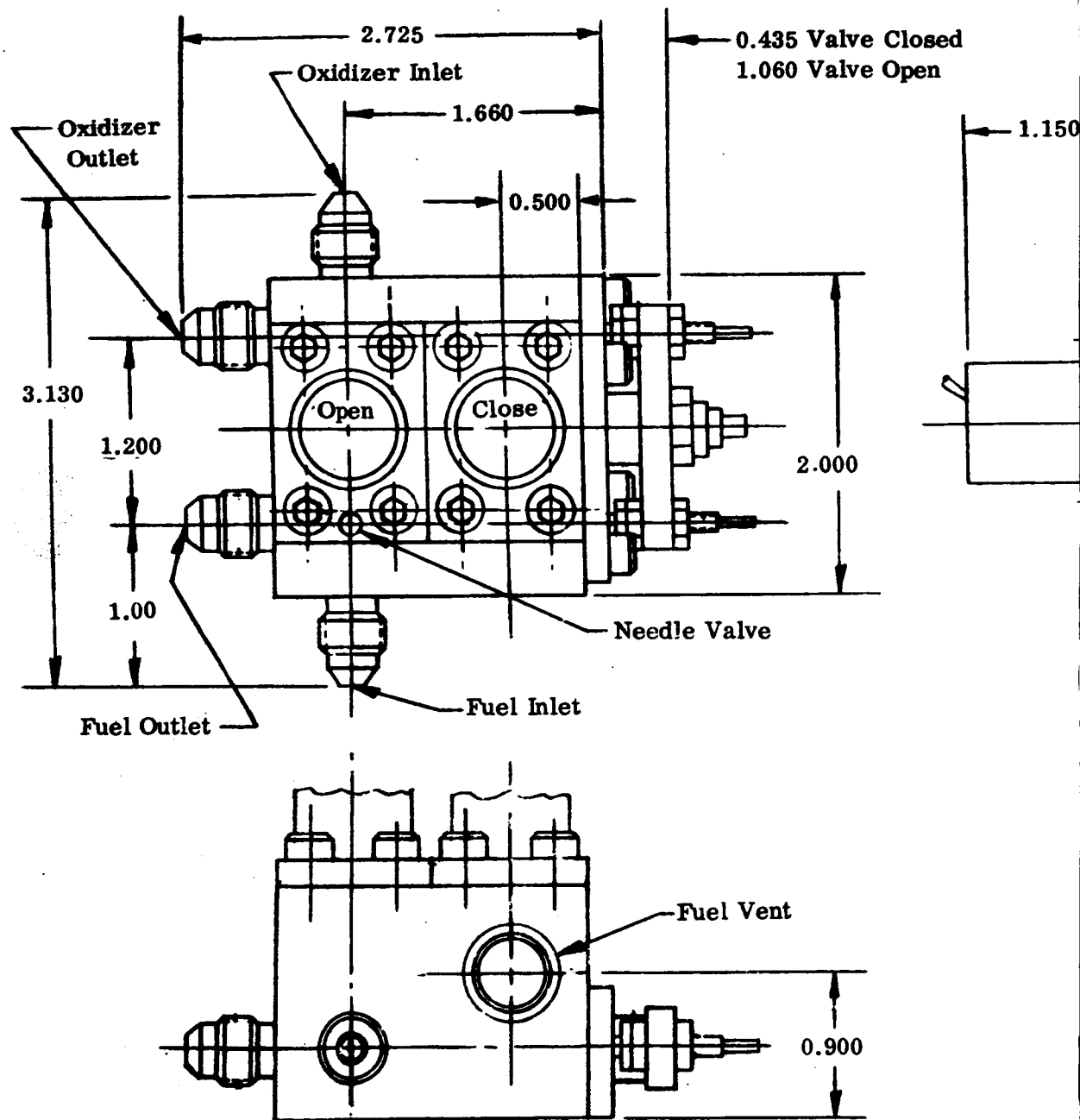
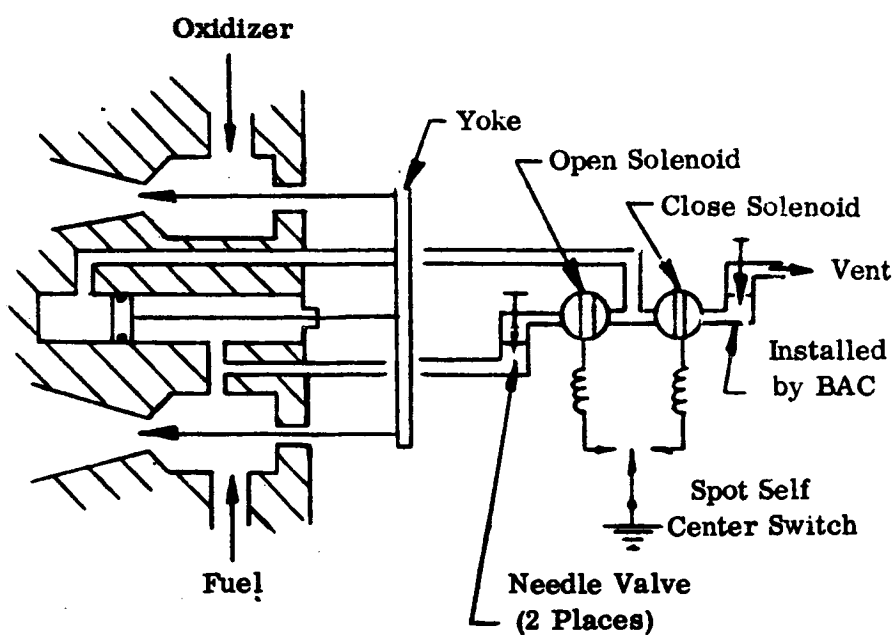
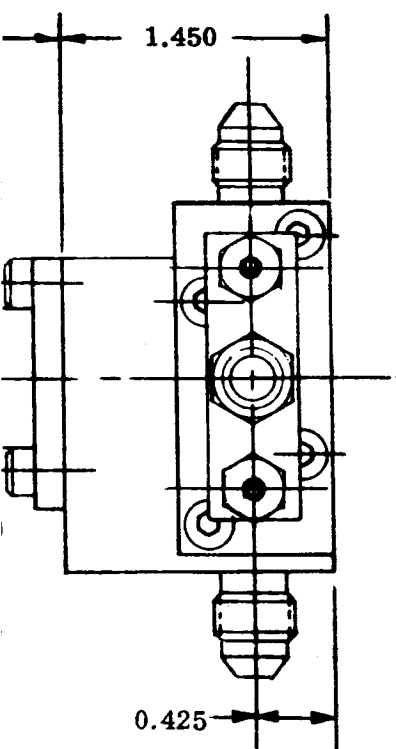


Figure V-190. Throttling Performance





Operational Schematic

Figure V-191. Bipropellant Valve (Fox Part No. 610646)

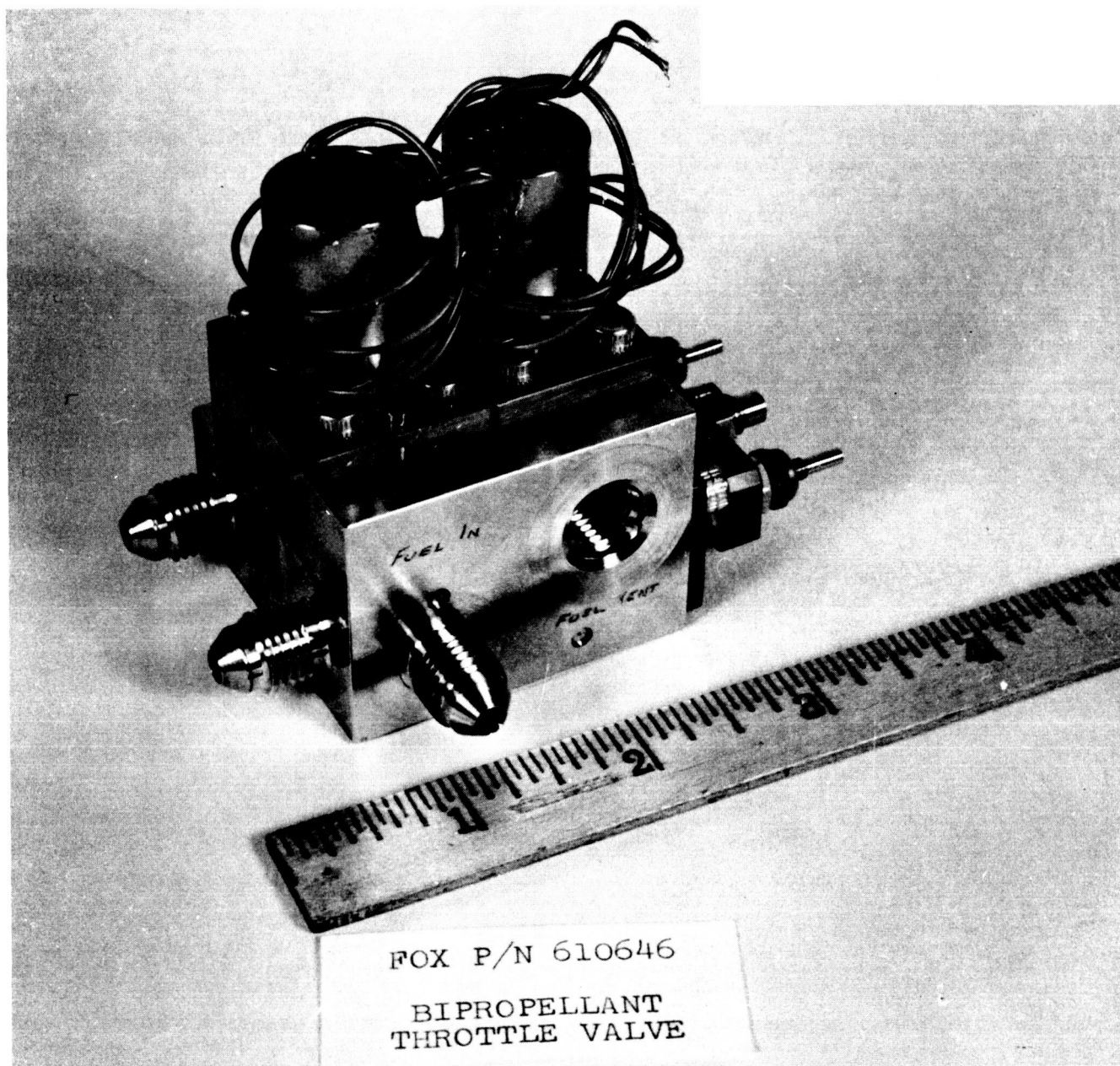
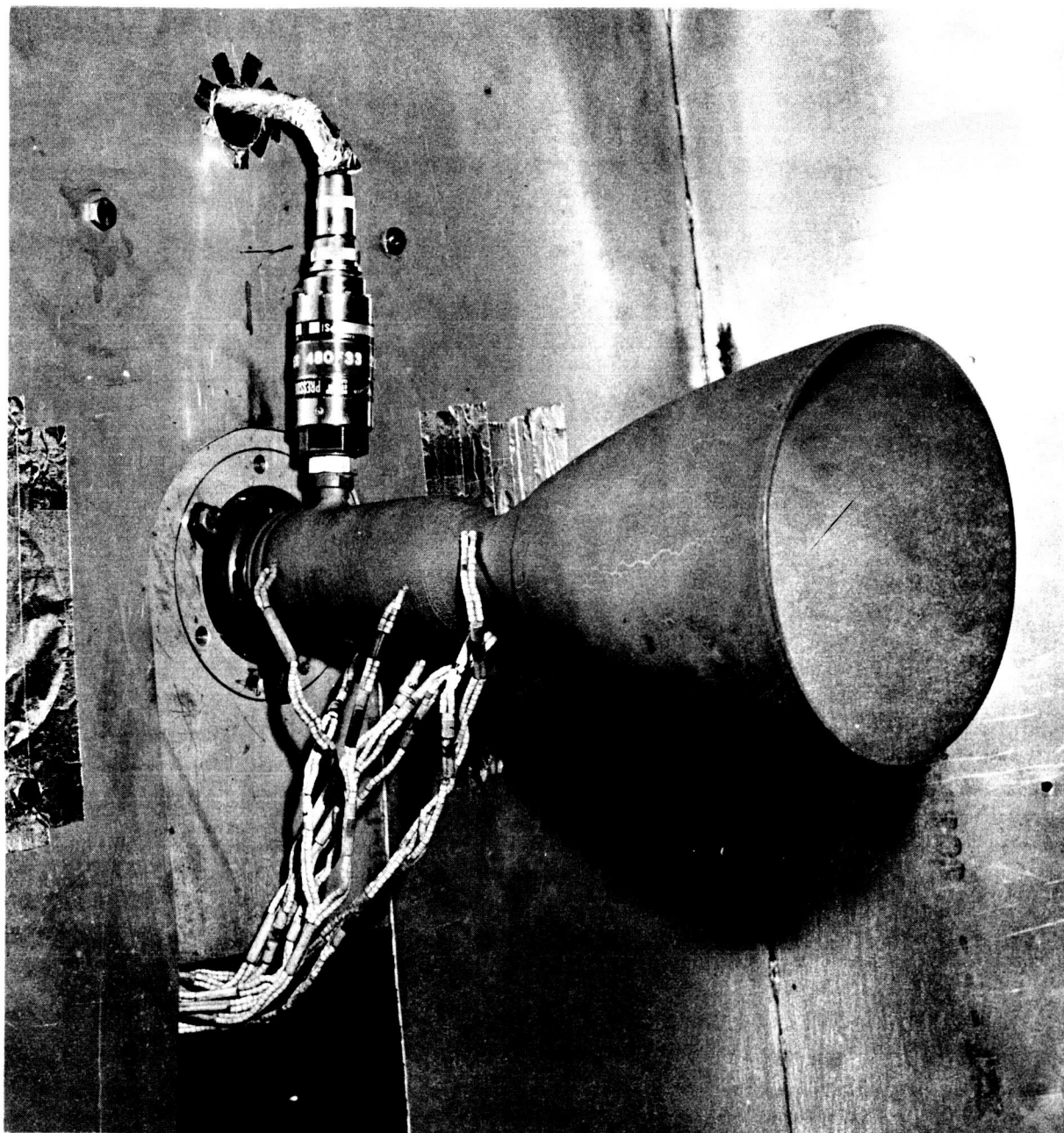


Figure V-192. Cavitating Venturi Bipropellant Valve

204157



203984

Figure V-193. Model 8414 Engine Test Installation - Thrust Chamber

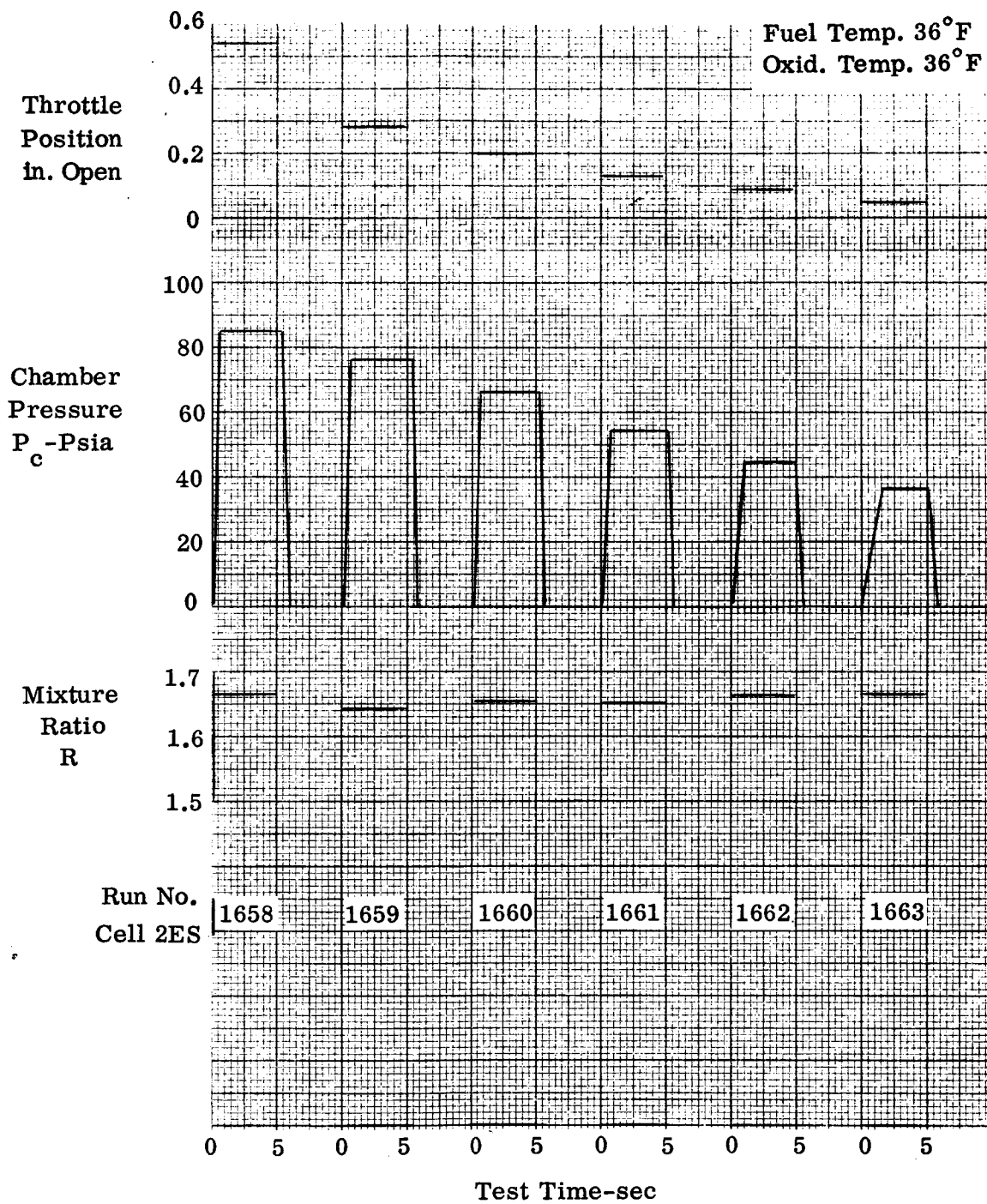


Figure V-194. Throttle Tests at Sea Level

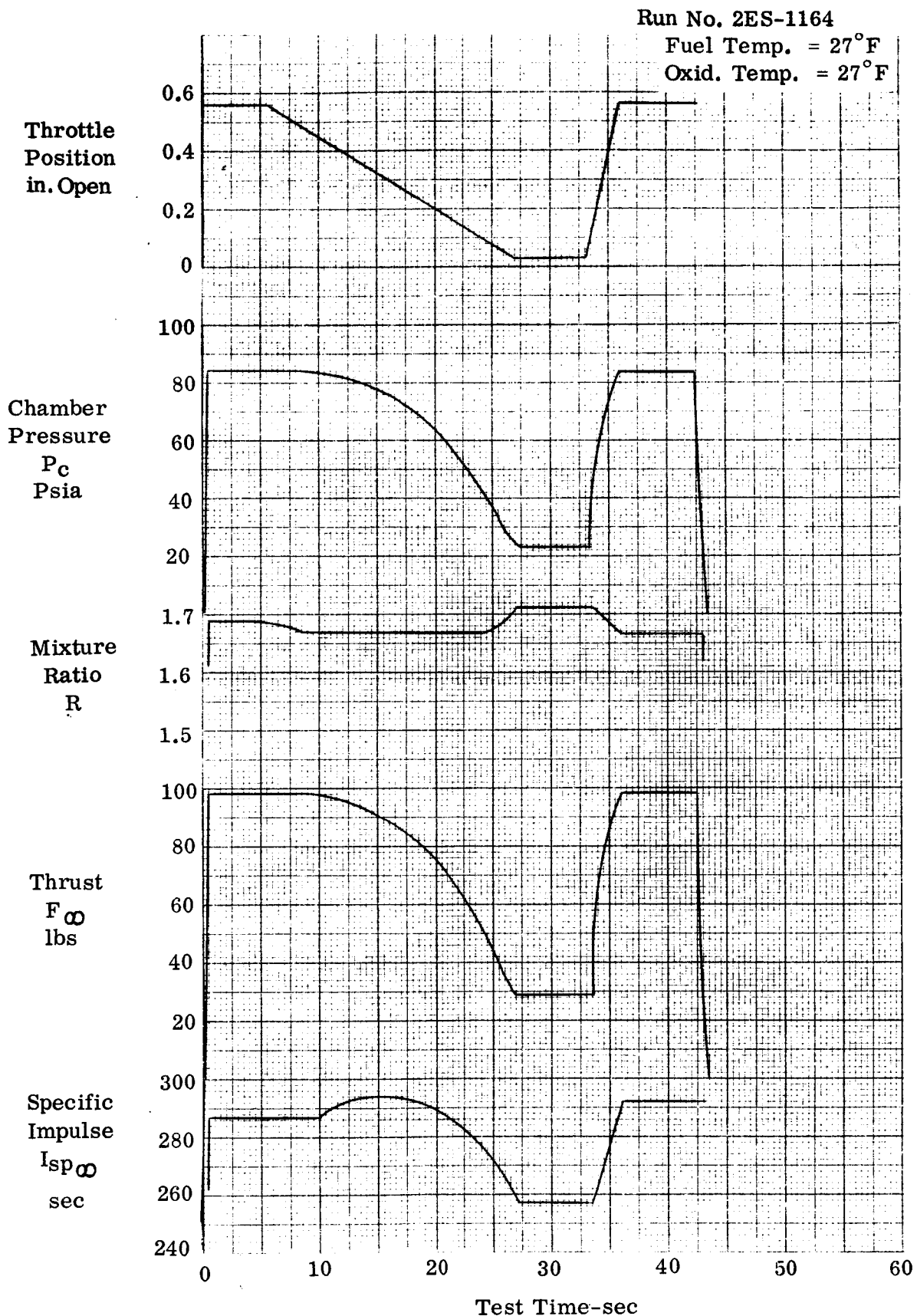


Figure V-195. Throttle Test at Pressure Altitude of 0.064 psia

Throttle
Position
in. Open

Run No. 2ES-1665

Fuel Temp. = 43°F

Oxid Temp. = 38°F

Chamber
Pressure
 P_c
Psia

Mixture
Ratio
R

Thrust
 F_{∞}
lbs

Specific
Impulse
 $I_{sp \infty}$
sec

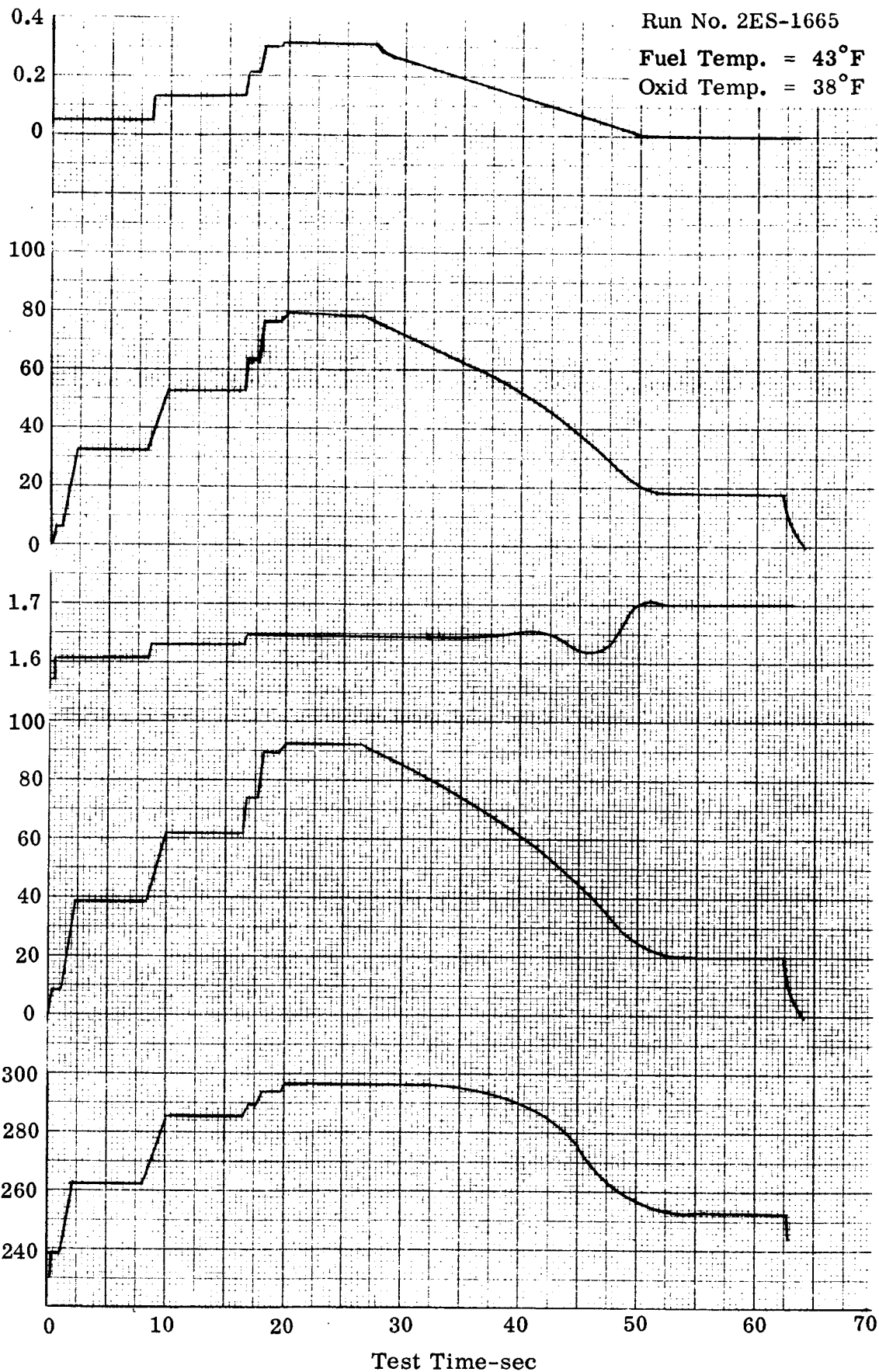
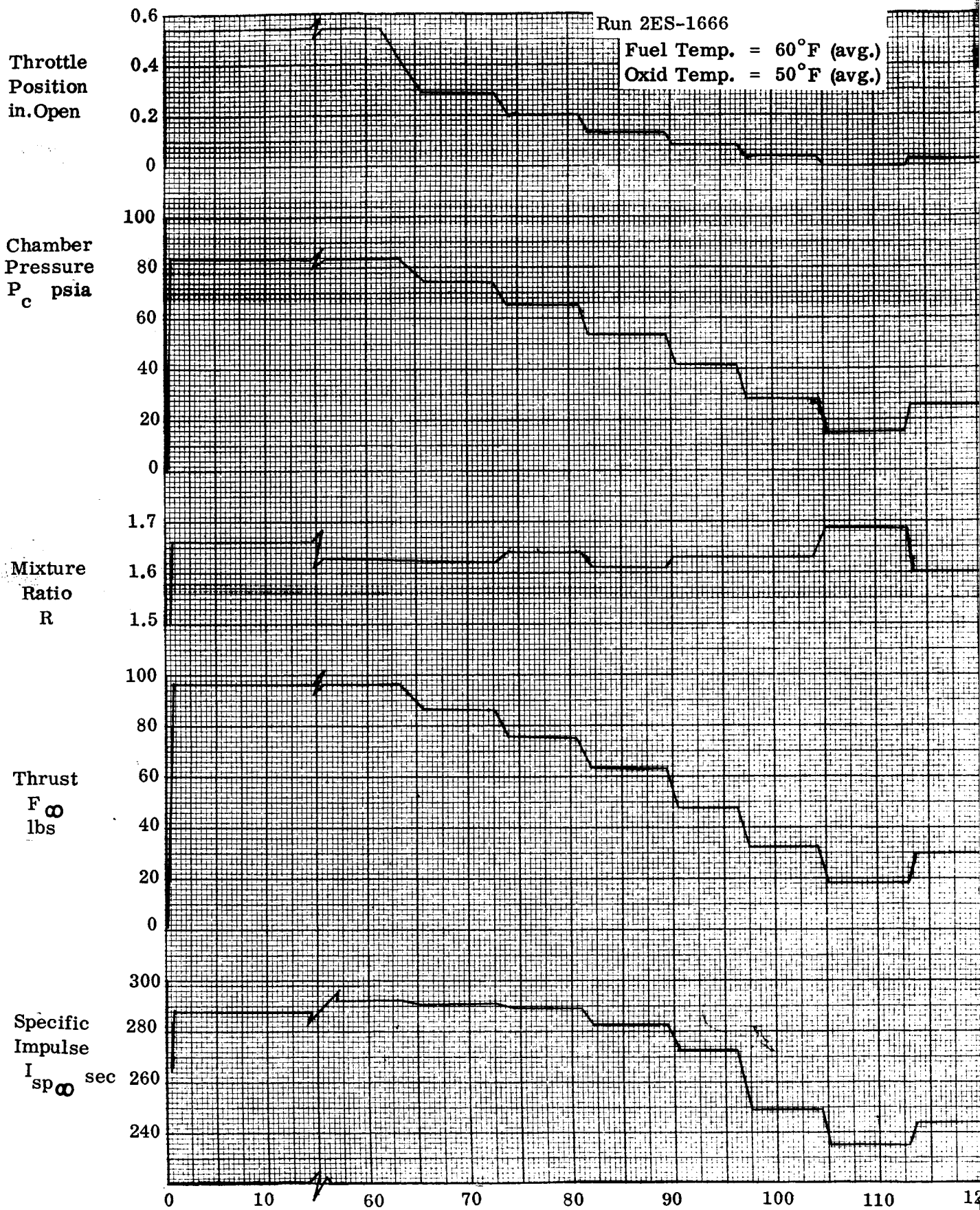


Figure V-196. Throttle Test at Pressure Altitude of 0.066 psia



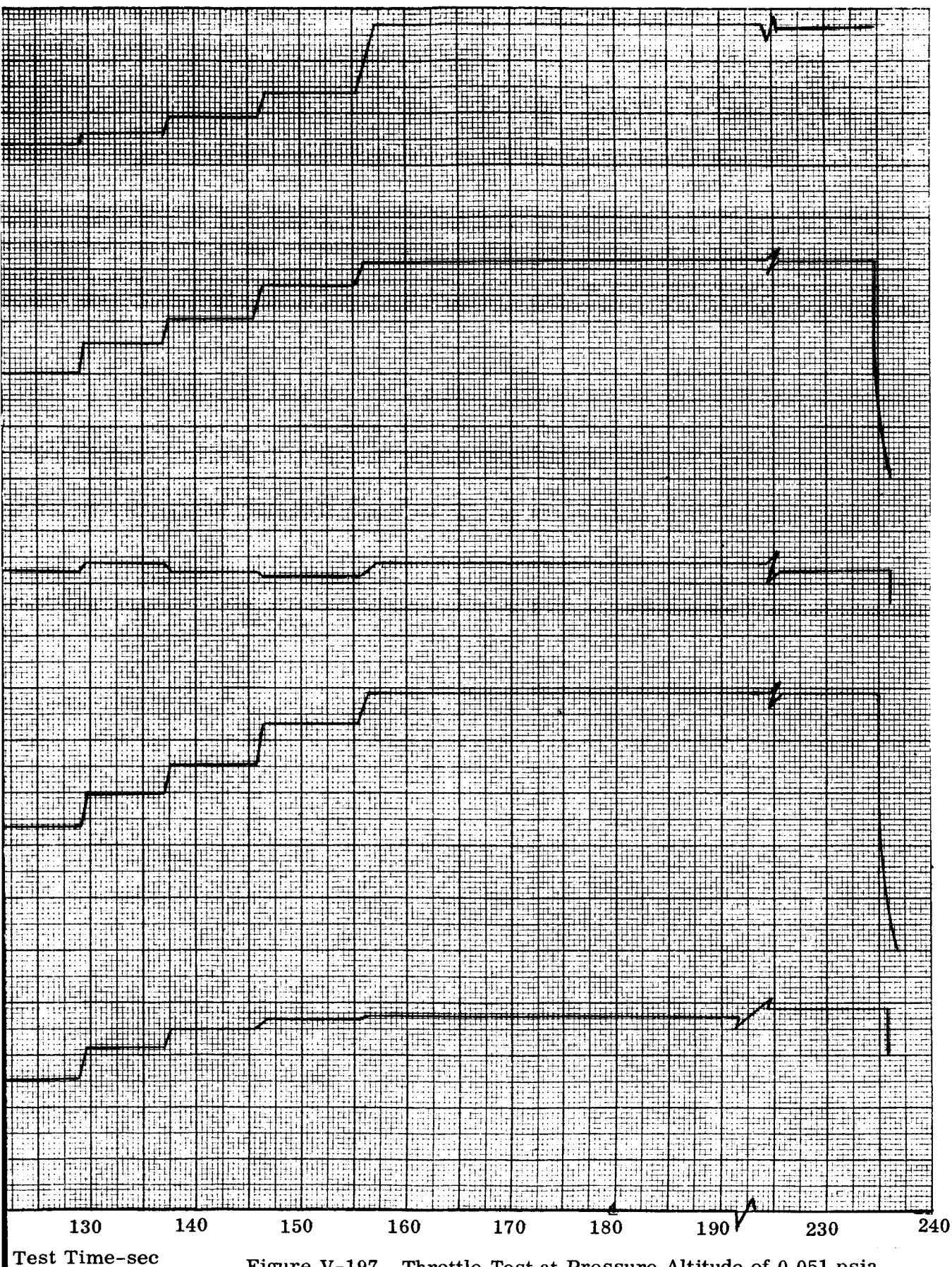


Figure V-197. Throttle Test at Pressure Altitude of 0.051 psia

V-345

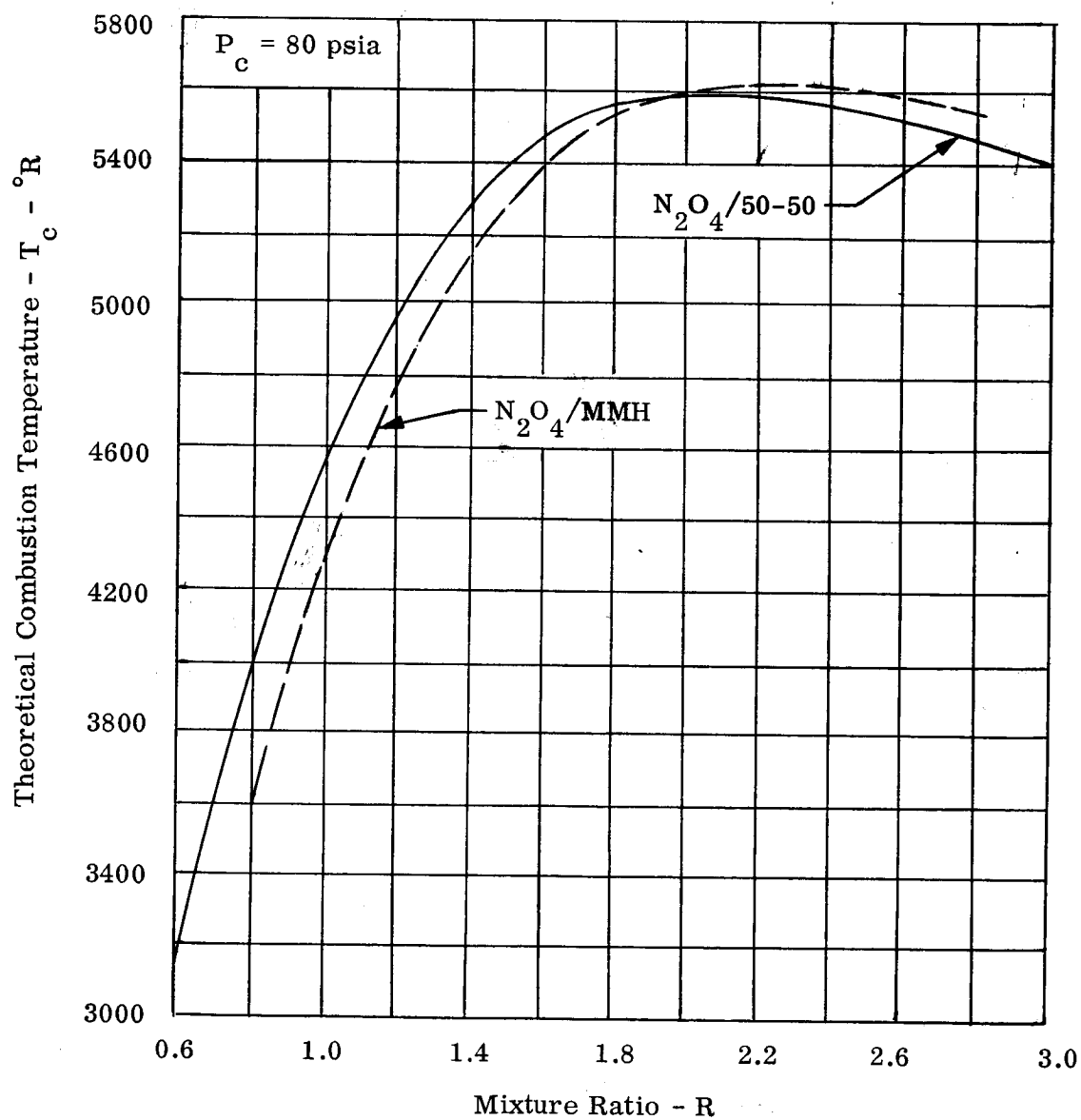


Figure V-198. Theoretical Combustion Temperature versus Mixture Ratio

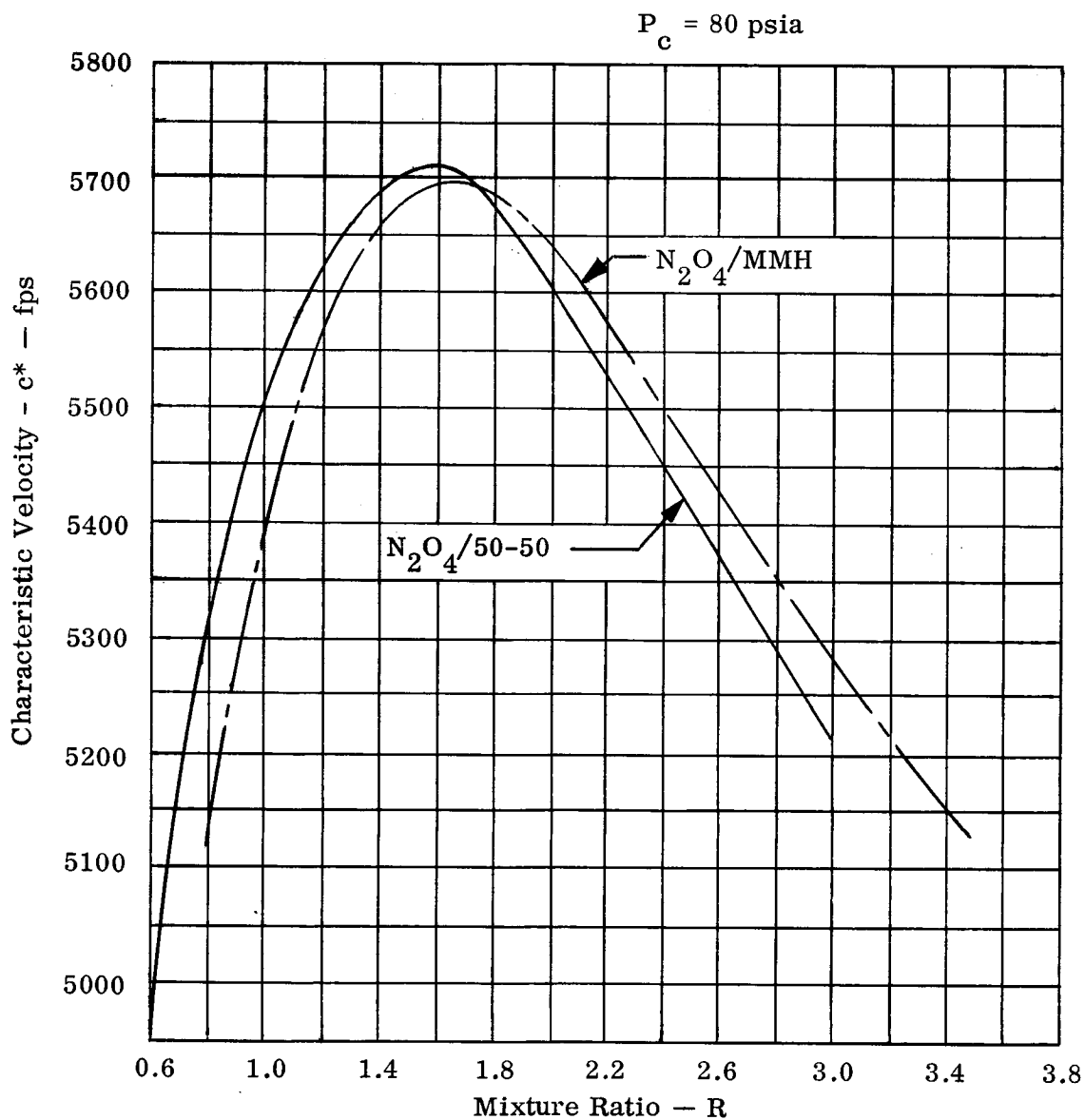


Figure V-199. Theoretical Characteristic Velocity versus Mixture Ratio

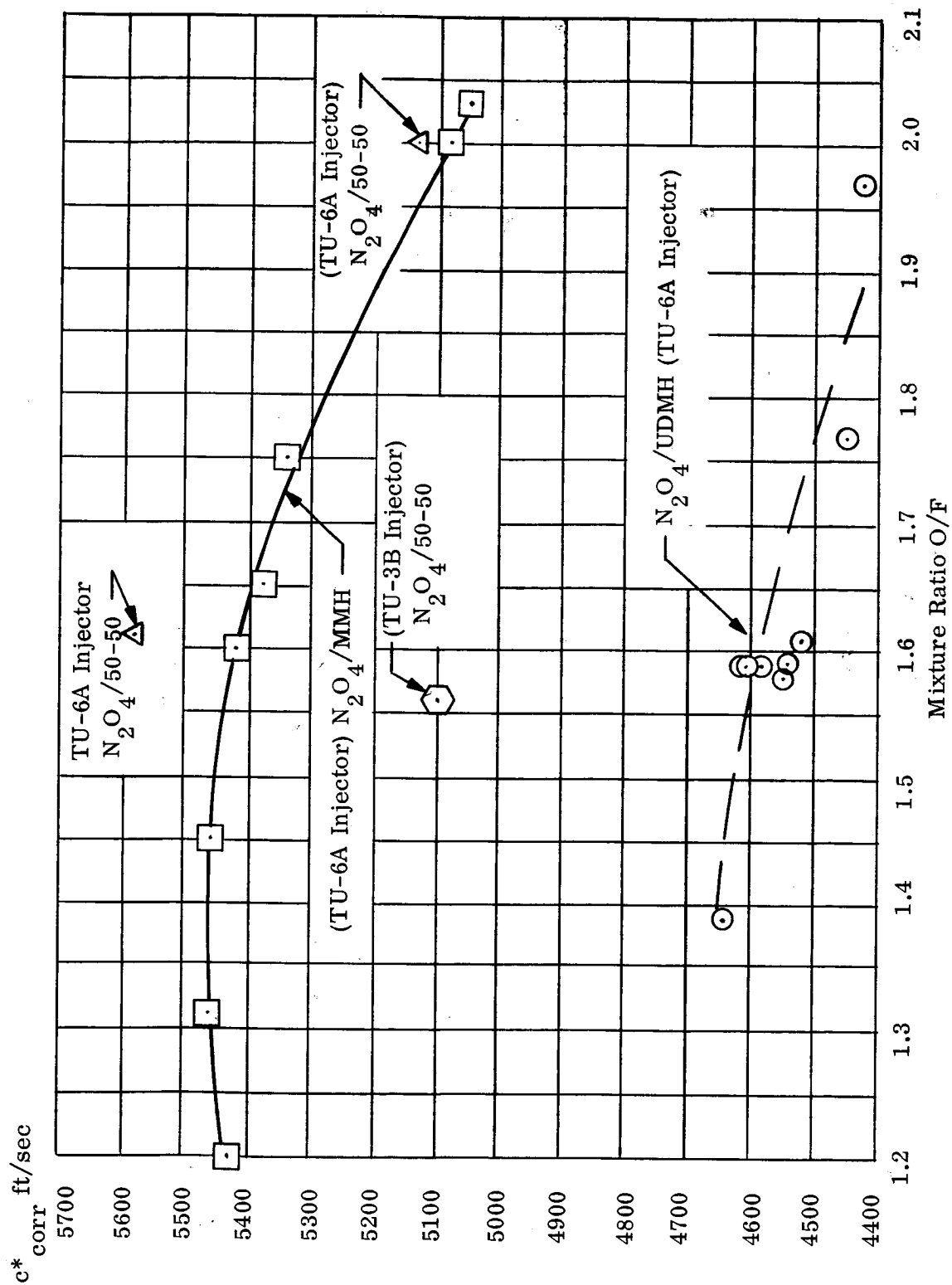


Figure V-200. Comparison of Performance Using N_2O_4 Oxidizer and MMH, 50-50, or UDMH Fuel
With Cb Chamber TU-6A and TU-3B Injector

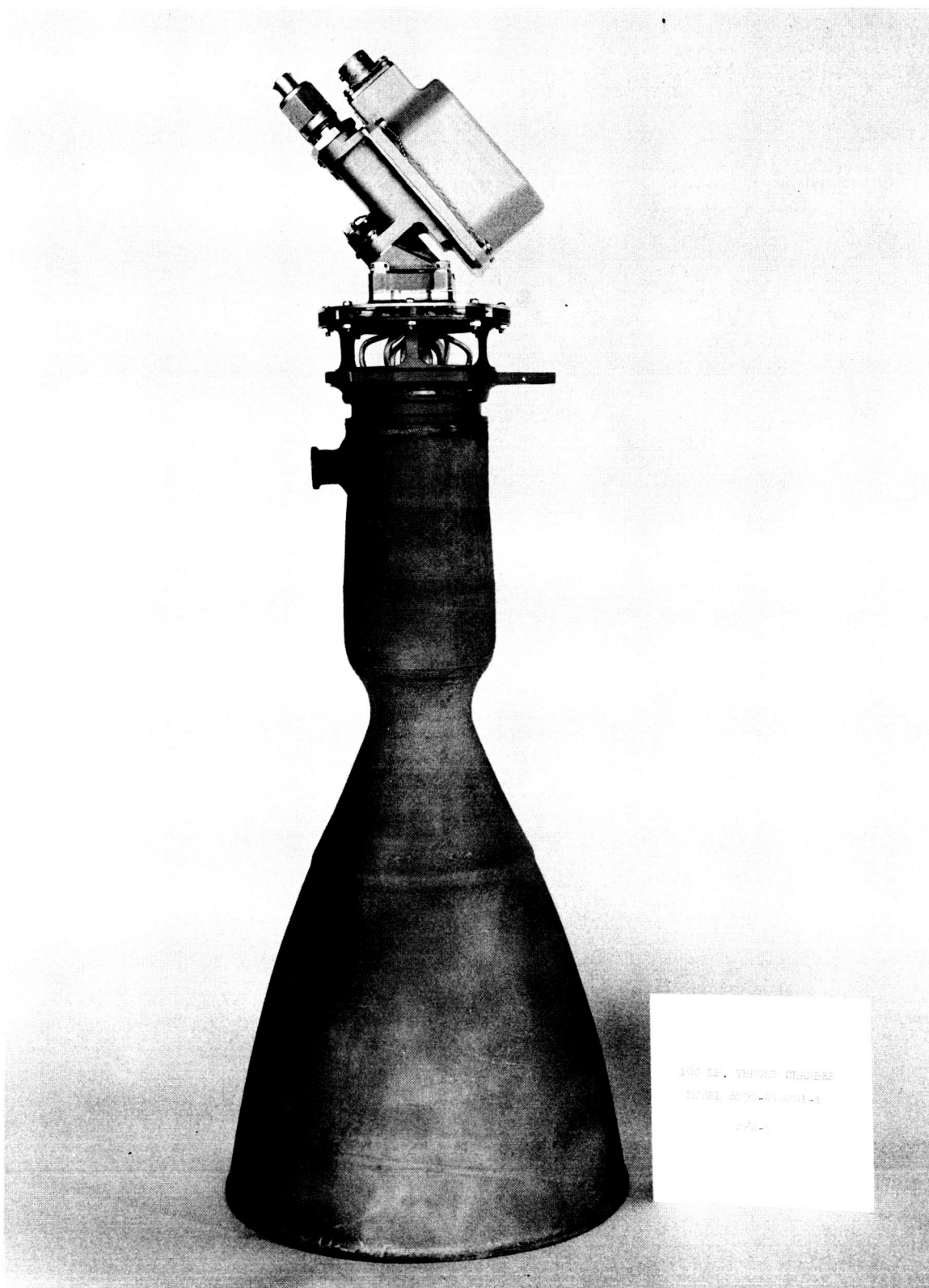


Figure V-201. 100-lb Thrust Chamber Model 8290-470001-1

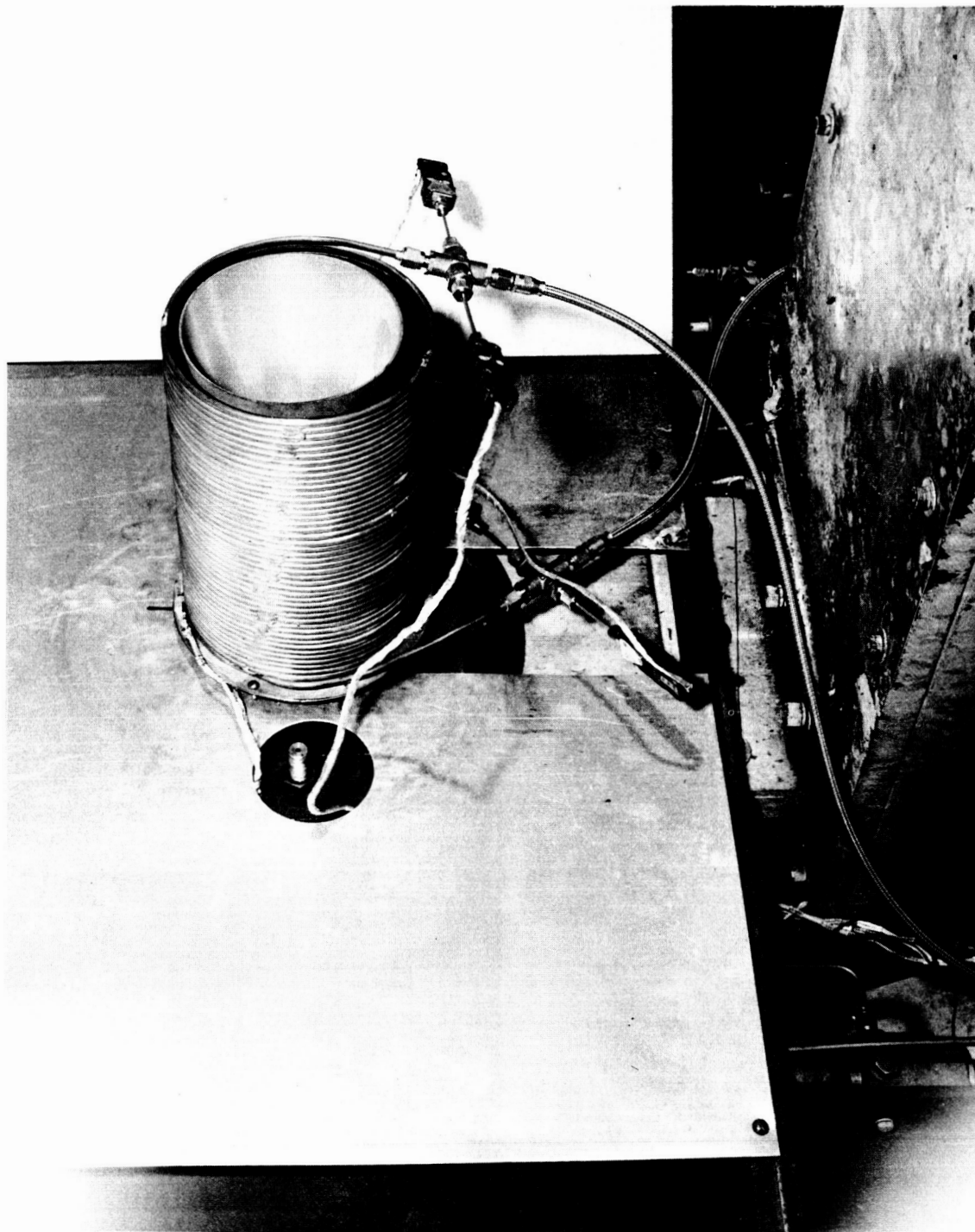


Figure V-202. Active Coolant Tests - 100-lb Thrust Chamber Assembly, Call 2ES

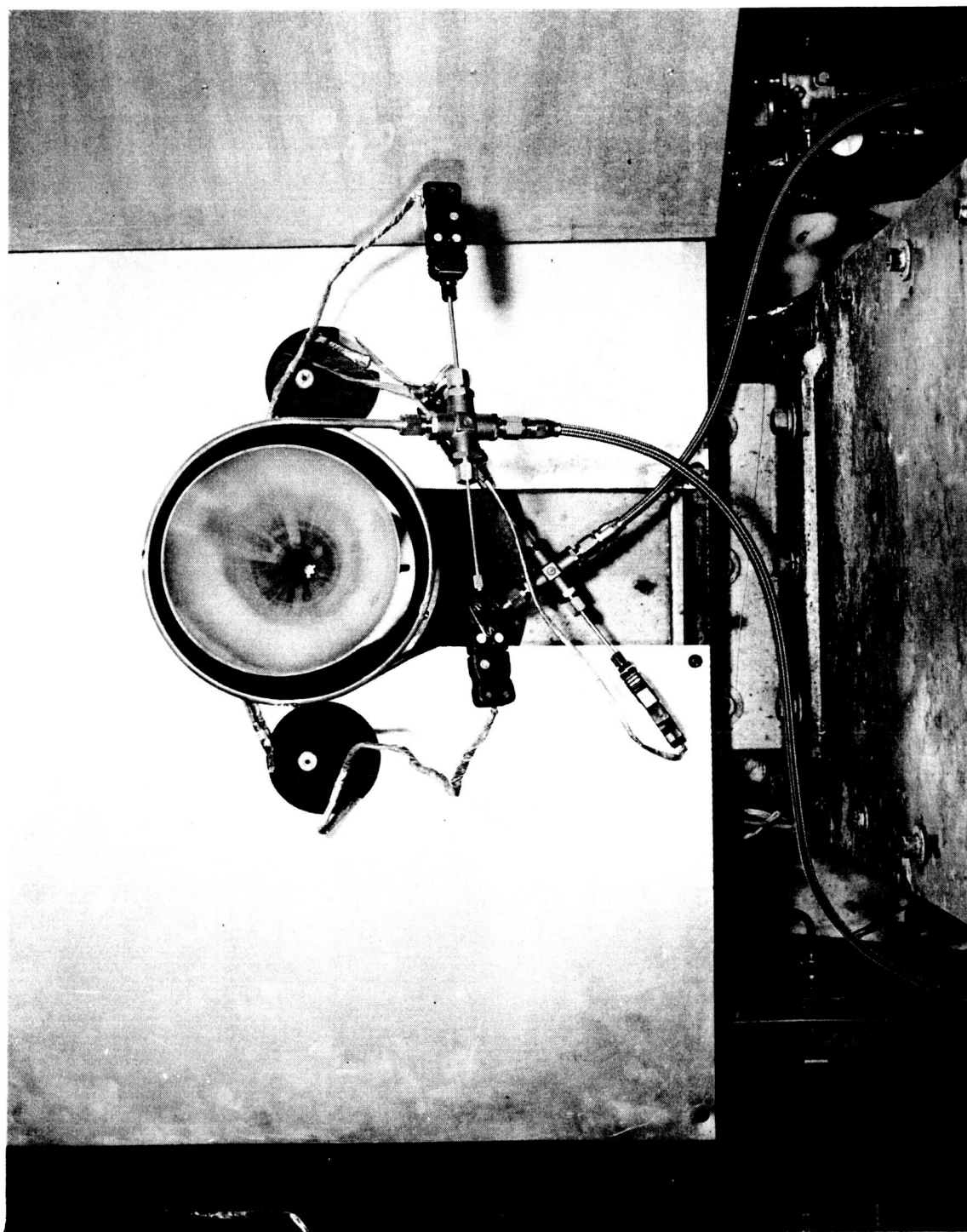


Figure V-203. Active Coolant Tests - 100-lb Thrust Chamber Assembly, Call 2K3



J. TEST CELL BUILDUP AND TEST STAND MODIFICATION

1. Sea Level Test Cell B-2

During the first quarter of this program, this sea level test cell was modified to allow testing of the various injection in the water cooled chamber, the stainless steel chambers, and the columbium alloy chambers.

This cell modification consisted mainly of providing additional instrumentation capability for the measurement of the test chamber temperatures.

2. Altitude Test Cell 2E-S

During the first and second quarter of this program the cell modification effort consisted of reworking the available integrated dynamic thrust and flow measurement test stand from the 5 pound thrust level to the 100 pound and providing additional instrumentation capability in the altitude test cell 2E-S.

This pulse mode thrust stand rework included redesign and fabrication of the following: the bellows tank assemblies, fast response propellant supply valves, the thrust calibration assembly, and the pulse control electronic circuitry. Two views of this test stand were shown in Section V-D. A propellant system assembly drawing of the stand is shown in Figure V-204.

Test firings were conducted on this modified test stand during the third quarter. Results of these tests showed rework to the gas bleed system on the bellows tanks was required to improve mixture ratio control and performance measurement. Modifications to the pulse control electronic circuitry was also necessary to allow greater pulse width variation.

Further modification to this test stand was not made due to cost.

During the last quarter of this program a simplified flexure type test stand was fabricated to measure thrust during the steady state and pulse mode series on the two demonstration engines.

