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APOLLO LUNAR MODULE DESCENT ENGINE
ABLATIVE CHAMBER - INJECTOR COMPATIBILITY
IMPROVEMENT STUDY

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I. INTRODUCTION AND SUMMARY

1.1 INTRODUCTION

This document is the final report covering an 18 month analytical and experimental technology program under contract NAS9-8229 with the NASA Manned Spacecraft Center to investigate and demonstrate material and design improvements that would reduce the ablative chamber throat erosion in the LM descent engine with: a) a resultant improvement in effective performance and, b) the ability to accommodate longer burn times with an increased margin of durability and reliability. The ultimate goal was to advance the capabilities of the LM engine should the need arise for some future mission.

At the time of contract award, there was no known future requirement. The LMDE had been qualified and was in production. Toward the end of the program, however, longer LM descent stage propellant tanks were proposed, effective with Apollo 16, and with the increased propellant came a new LMDE requirement called an extended G mission duty cycle. It was demonstrated that the qualified LMDE chamber could survive the increased firing time, but that throat erosion would increase by 50-to-100 percent over that experienced during the Qualification B Program. Such erosion was considered to be unacceptable for the mission.

When the new need arose, the improved prototype chamber that was the fruition of this program was being fabricated. The improvements were incorporated concurrently into a Qualification B chamber version and both chambers were test fired successfully to the new extended G mission duty cycle, with less than half the erosion exhibited in a qualified chamber fired similarly. All three tests were conducted with the same head end assembly. The improved LMDE chamber is now undergoing Design Verification Testing.

The results of the program demonstrate the desirability of low level technology programs, even on production components. Such programs serve the twofold purpose of maintaining: a) the technical proficiency of a developmental staff, and b) the ability to incorporate developed improvements rapidly into the production item should more stringent requirements arise.

1.1.1 Mainline Design

The mainline Lunar Module Descent Engine Thrust Chamber Assembly, Figure 1-1, consists of a composite ablative-cooled combustion chamber, ablative-cooled expansion cone to an area ratio of 16:1, a crushable radiation-cooled nozzle extension to an area ratio of 47.4:1, and a gimbal ring assembly. The ablative components are encased in a continuous titanium case surrounded by a stainless steel foil/glass wool composite thermal blanket. The radiation-cooled nozzle extension is made from a columbium alloy and contains a high-emissivity, oxidation-resistant coating. The flanged chamber and nozzle extension are bolted together. Joint leakage is prevented by an Inconel X-705 K-seal. The gimbal assembly is located at the chamber throat plane and provides for a ± 6 -degree gimbal of the thrust vector during flight.

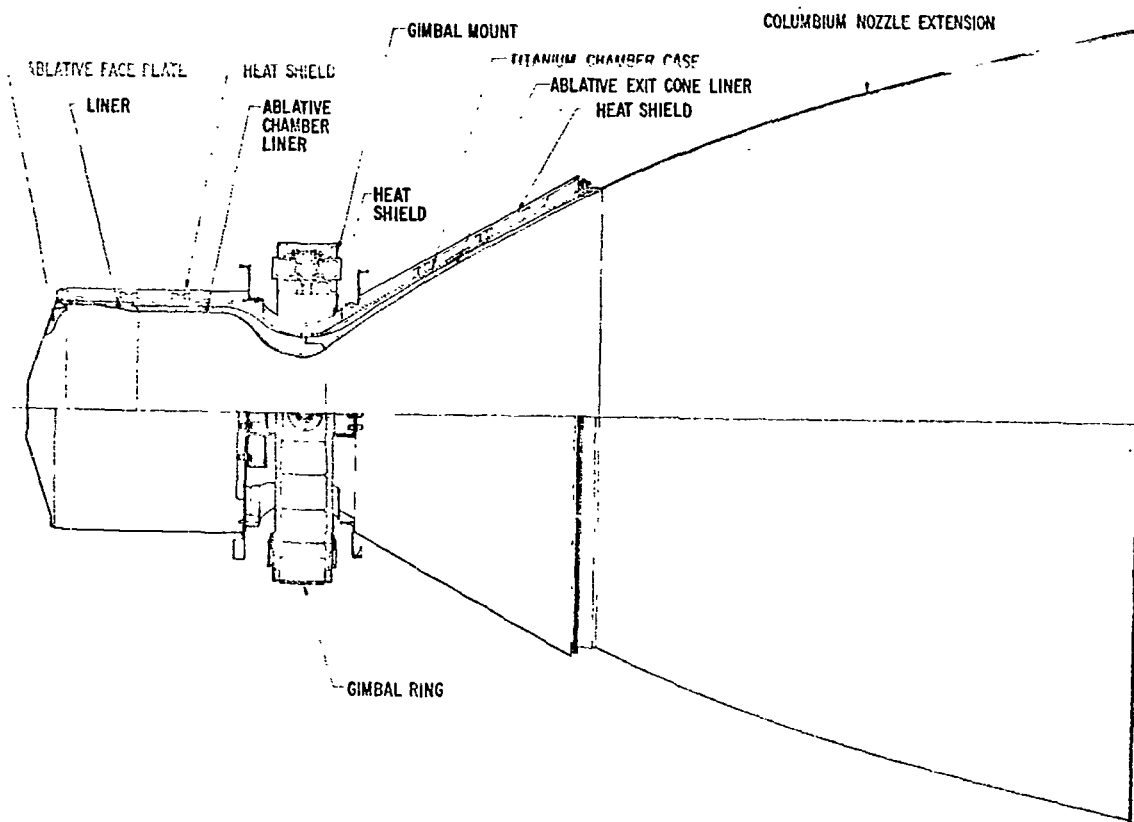


Figure 1-1. LMDE Thrust Chamber Assembly

Figure 1-2 depicts the chamber configuration that successfully passed the Phase B Engine Qualification. It has proved to be particularly efficient in meeting present performance requirements. The ablative liner is designed to char completely through. Liner thickness is sized to maintain the 6Al-4V titanium case temperature within reasonable limits at engine shutdown. The titanium case is the principal pressure vessel and is capable of operating at specification safety factors at temperatures in excess of 800°F. The attachment flanges for the injector, gimbal, and nozzle extension are machined as an integral part of the titanium case. Aside from its good high-temperature strength, titanium is also lightweight and relatively easy to machine.

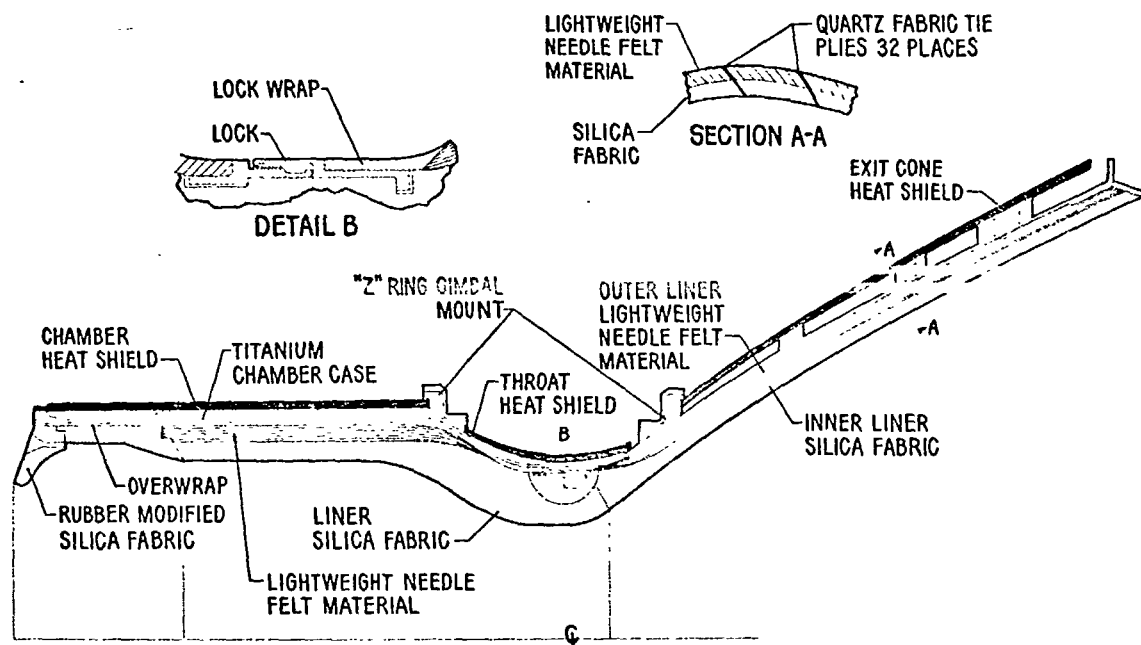


Figure 1-2. Qualified Chamber Liner

The use of a continuous metal case requires a two-piece ablative liner with insertion of each portion from either end of the case during assembly. The two are joined at the throat by a positive lock which, after engagement, cannot be separated without complete destruction. Development tests have shown this lock to perform satisfactorily with more than adequate strength to survive the vibration design loads. When

the engine is operating, the lock is redundant since the net thrust holds the exit cone liner in the case. Additional retention is supplied by the nozzle extension at the $c = 16$ location.

The back wall or external wall temperature requirements are met by a composite thermal blanket (Figure 1-2) composed of a fiberglass wool enclosed by two sheets of 0.0015-inch thick stainless steel foil. This heat shield provides a lightweight (0.25 lb/ft^2), efficient, thermal system that protects the vehicle engine cavity from being exposed to temperatures above 400°F . In addition to the high thermal efficiency of the blanket, the design margin is increased by control of the surface emissivity of the foil and the titanium case.

Actual throat erosion experienced during nominal duty cycle firings in Phase B Engine Qualification Testing is summarized in Table 1-1.

Table 1-1. Qualification Nominal Duty Cycle Calculated Erosion
Summary - Percent Increase in Throat Area

Duty Cycle Phase	Engine No.			
	4	5	6	7
FTP Phase	8.8	7.1	4	8.75
58 to 68 Percent Thrust Ramp	0	0	0	0.75
60 to 40 Percent Thrust Phase	0	0	1	0
25 Percent Thrust Phase	14.4	8.8	10.6	9.9
Total Erosion at Depletion of Vehicle Propellant Load	21.2	14.4	13.9	17
Total Erosion for Entire Duty Cycle	23.2	16.8	15.9	19.4

1.1.2 Mainline Duty Cycle

The nominal LMDE duty cycle is shown in Figure 1-3. It consists of a 33-second orbit-injection firing, a one-hour coast, and a nominal 784-second descent firing. Both firings start and remain for 26 seconds at low thrust while the vehicle is stabilized. During the orbit-injection firing, the engine is throttled to full thrust, nominally 9850 pounds, for 7 seconds and then shut down. During the main descent firing, the engine is again throttled to full thrust after the startup; this braking phase lasts from 320 to 370 seconds at full thrust and is completed in an additional 68-to-127-second period at about 6000 pounds thrust. In the next 115 seconds, a vehicle flare-out is accomplished, with throttling in the range between 6000 and 4000 pounds thrust. Subsequent engine operation involves gradual throttling near 2500 pounds thrust as the vehicle hovers. Finally, the relatively rapid-throttling touchdown maneuver is accomplished. The total firing time required of the engine is 1000 seconds, which includes 90 seconds of acceptance testing plus a total of 910 seconds of duty cycle firings. This duty cycle time includes the capability of consuming what remains of the full vehicle propellant load of 17,930 pounds during the hover phase.

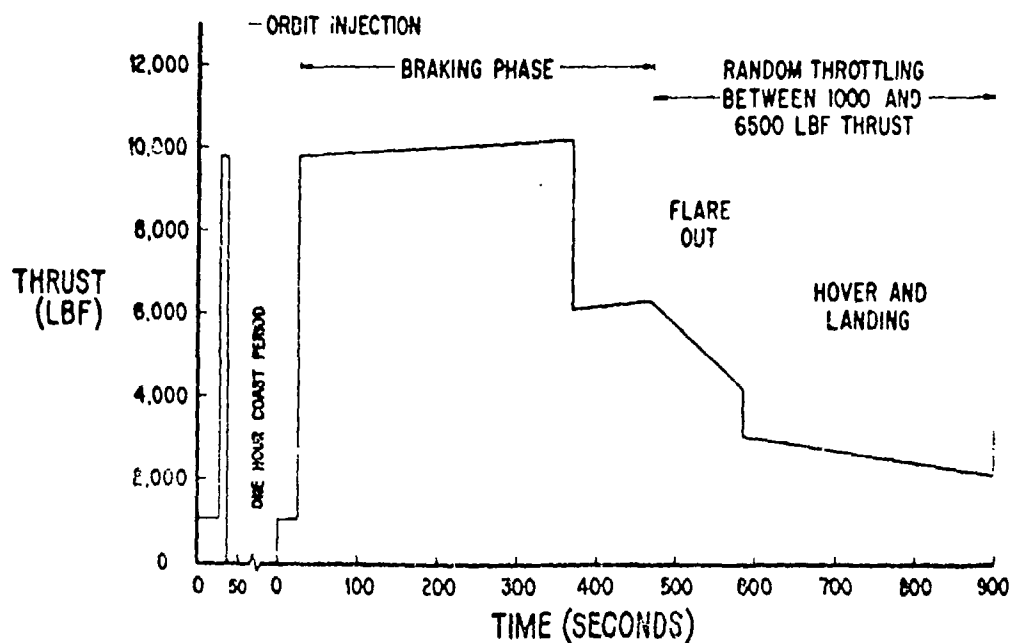


Figure 1-3. Nominal LMDE Duty Cycle

The lunar module gas-pressurized feed system provides propellants at the engine interface at 220 psia during full thrust operation. As the engine is throttled, the interface pressures approach the feed system tank pressures, which are controlled to 235 ± 2 psia. The propellants are nitrogen tetroxide (N_2O_4) and 50/50 (N_2H_4 /UDMH) at a 1.6 mixture ratio.

At the full thrust throttle position, the nominal head end chamber pressure is 104 psia, throttling down to 11.4 psia at the ten-percent thrust level.

1.1.3 Extended Mission "G" Duty Cycle Requirement

During the latter phase of the CIC Program, a definitive extended mission for the LM engine arose. The mission called for use of 19,562 lb of propellant in the duty cycle of Figure 5-34. This duty cycle was employed for the final demonstration firings of this technology program.

1.2 SUMMARY

Major accomplishments of the CIC Program were as follows:

- Detailed analytical and experimental evaluation of the current descent engine chamber wall environment and the correlation of this environment with the resultant thrust chamber overall durability/throat erosion.
- Determination of variations in thermal environment as functions of internal chamber contour, injector flow distribution, barrier coolant flow, and engine thrust level.
- Analytical evaluation and experimental confirmation of improved ablative chamber materials and chamber designs.
- Verification testing of an improved descent engine thrust chamber design (internal contour and material) in a flightweight configuration, with a significant reduction in throat erosion even though the duty cycle duration was increased by 10 percent.
- Incorporation of the improved chamber liner material into the mainline LMDE chamber (Qual B design) with a significant reduction in throat erosion over the extended duty cycle.

Improvements in initial performance and reduction in throat erosion brought about by contour changes were not marked enough to warrant change in the mainline design. However, the introduction of quartz-fabric-

reinforced phenolic resulted in a significant improvement in erosion resistance and it has since been incorporated into the online design. Design verification testing of the redesigned chamber is now in progress under the prime LMDE contract.

As defined by the contract, the program was conducted in three phases, the first two concurrently, consisting of the following:

Phase I - Define/Characterize the Internal Combustion Chamber Environment.

Experimental heat transfer studies were conducted with available qualification design head end assemblies to evaluate the thermal environment to which the ablative thrust chamber was subjected. Specially designed and instrumented thrust chambers were fabricated for obtaining the desired chamber wall environment data. The basic objective of these tests was to define the local heat transfer conditions at the chamber wall and correlate these data with the overall ablative chamber durability/throat erosion.

Phase II - Evaluation of Improved Ablative Chamber Materials and Designs.

A literature and industry survey was made to determine candidate ablative materials for an advanced ablative chamber design. A laboratory investigation showed that by the proper balance of carbon and silica in an ablative structure favorable endothermic reactions could be induced resulting in the formation of stable refractory SiC instead of the volatile SiO product. The reaction mechanism was duplicated in specimens exposed in an inexpensive torch simulation of the LMDE combustion products. Other candidate ablative and hard throat materials were evaluated similarly.

The most promising advanced ablative materials were fabricated into a composite ablative throat and test fired at sea level in a chamber contour (Contour C) that promised improved engine performance. As a result of this firing, quartz-fabric-reinforced phenolic was chosen as the material to be used in a subsequent flightweight chamber test firing. A standard laminated LMDE silica-fabric/phenolic chamber was built to the configuration (Contour A) that exhibited the least heat flux and gas recovery temperatures at FTP and test fired at sea level with the qualification configured injector. From these tests, an advanced chamber

design was recommended for NASA/MSC approval and subsequent fabrication of a flightweight unit in Phase III.

A hard throat design concept was analyzed thermally and presented to NASA/MSC.

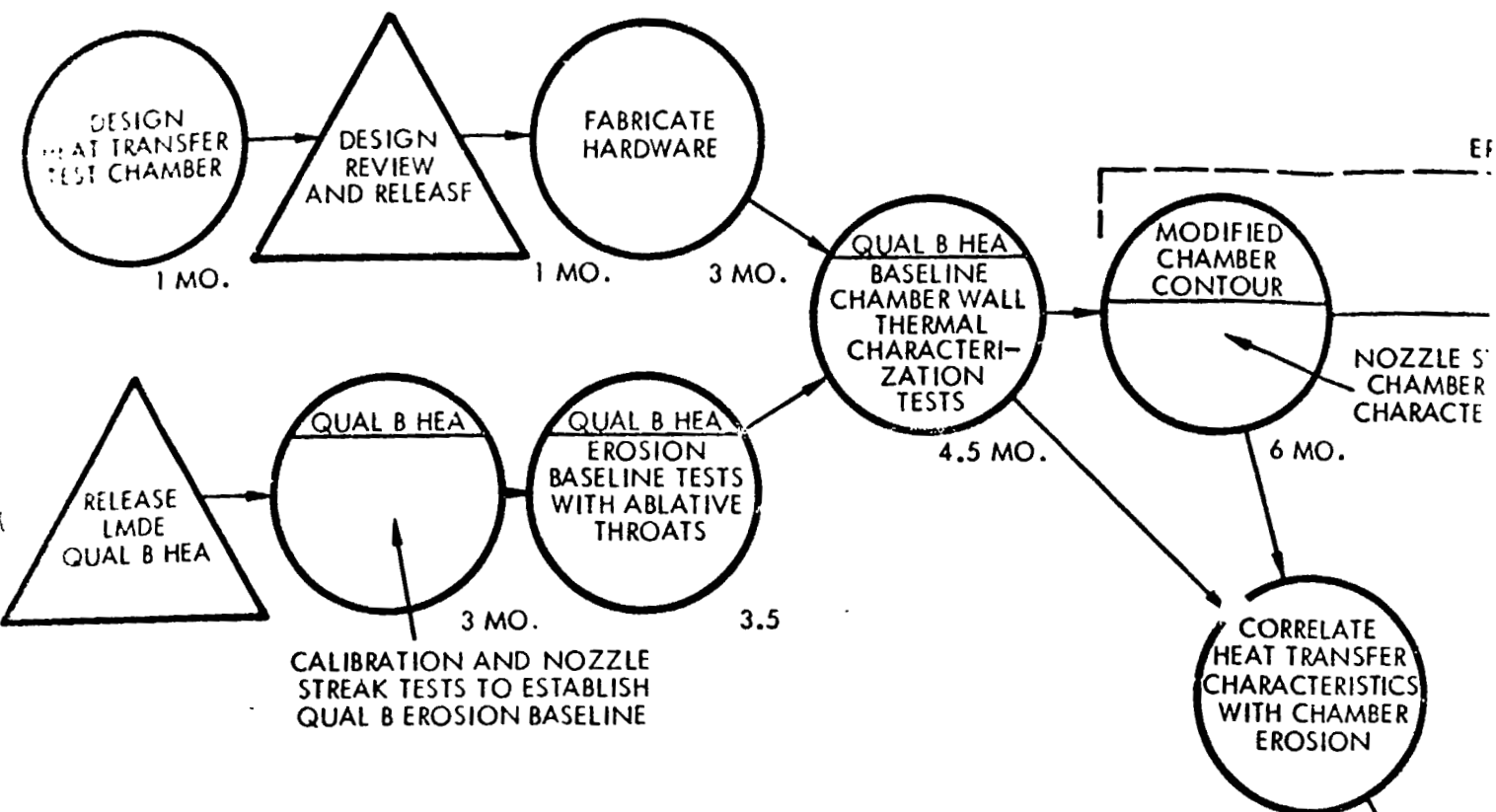
Phase III - Design, Fabrication, and Test of an Improved Descent Engine Thrust Chamber.

A flightweight prototype chamber was fabricated by production procedures. The improved chamber design (Contour C) was verified in an extended mission duty cycle altitude test, using a qualification-type head end assembly of defined ablative chamber compatibility characteristics. The head end assembly utilized for this testing was typical, from a chamber erosion standpoint, of the Qual B Descent Engine configuration. Although the promised performance improvement was not attained, throat erosion was reduced significantly.

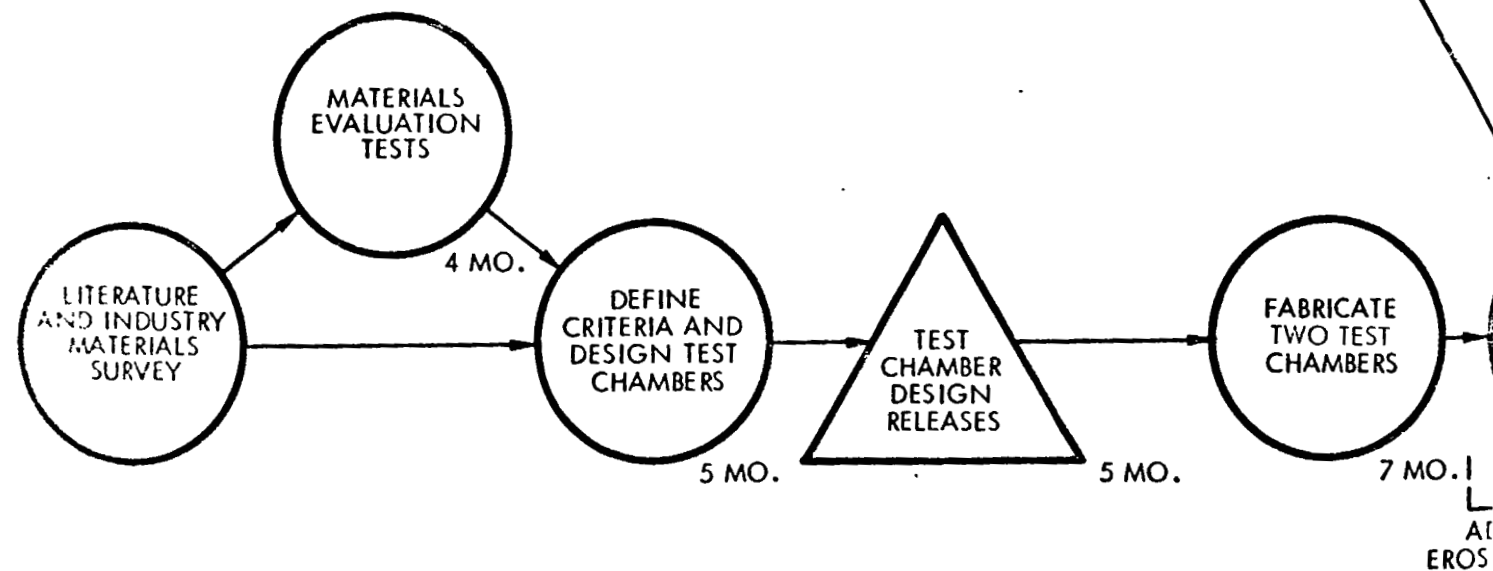
While the improved chamber, described above, was being fabricated, a new requirement, called an extended G mission duty cycle, was placed upon LMDE for Apollo 16 and subsequent missions. It was demonstrated that the LMDE chamber could survive the increased firing time, but erosion would increase by 50 percent over that experienced in the Qualification B Program. When the LMDE chamber was redesigned to incorporate the quartz-fabric/phenolic and other improvements resulting from this program, however, it survived the new duty cycle with less than half the erosion of the mainline silica chamber.

The overall program logic is shown in Figure 1-4. The test matrix is shown in Figure 1-5.

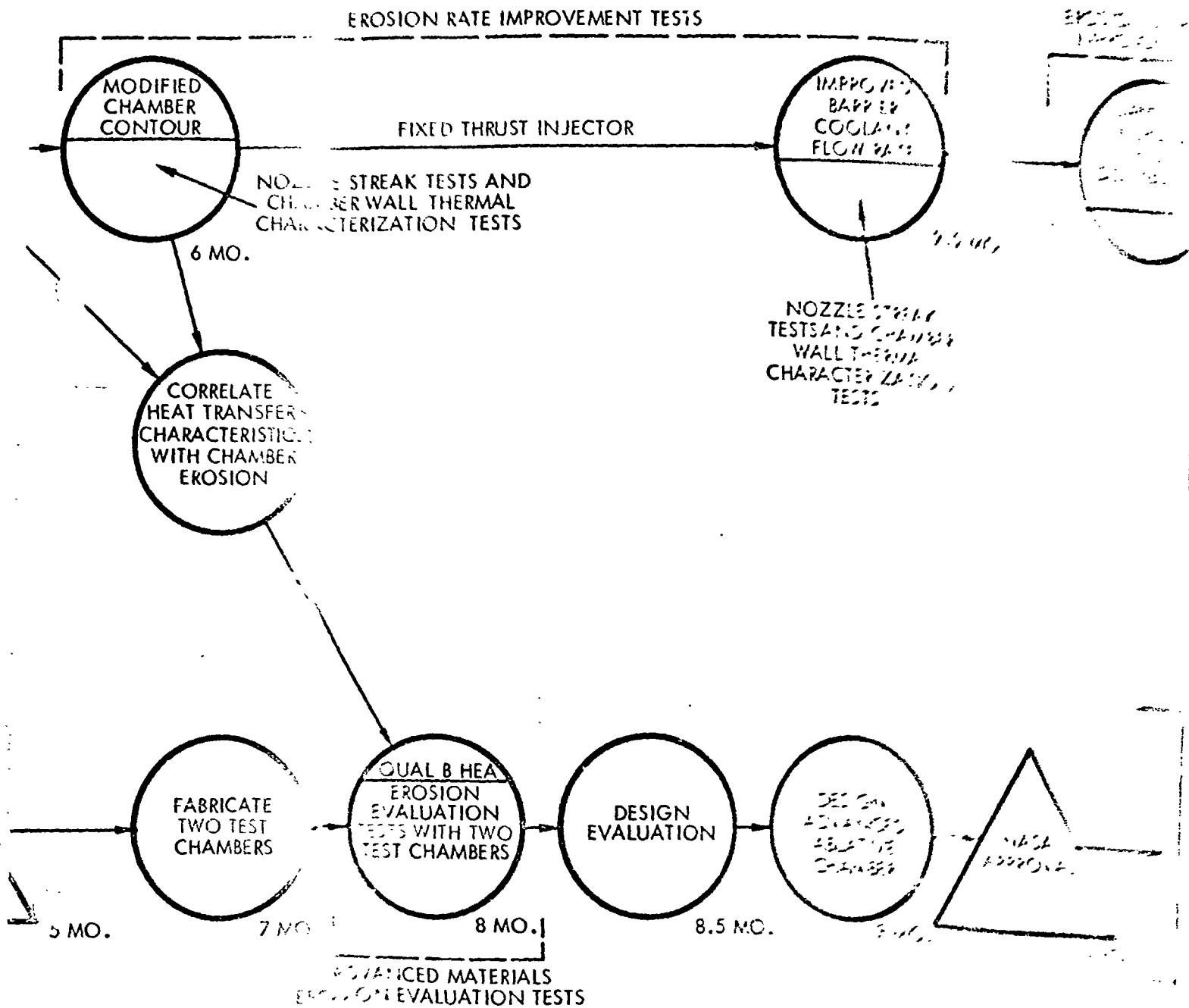
PHASE I



PHASE II



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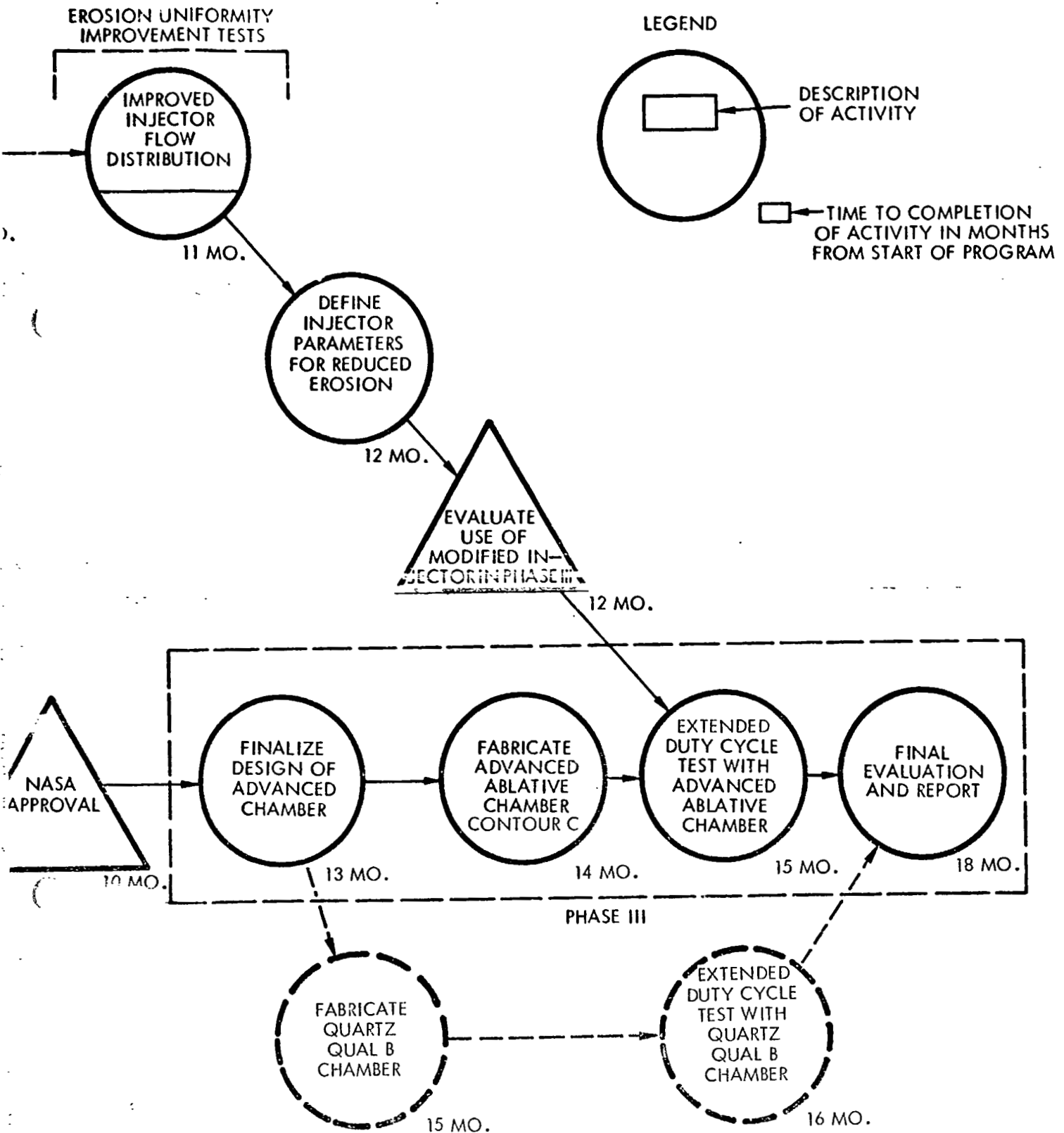


Figure 1-4. Overall Program Logic Flow Chart

FOLDOUT FRAME 3

TEST OBJECTIVES	TYPE OF TEST										TEST CONDITIONS						HARDWARE CONFIGURATION										FACILITY																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																								
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											THRUST SETTING* (% OF 10,500 LB)	NUMBER OF REACTIVE TESTS	DURATION OF EACH TEST	TOTAL DURATION PER SERIES	STANDARD LMDE WATER COOLED	FIBER GLASS NOZZLE		ABLATIVE NOZZLE	LMDE ABLATIVE NOZZLE	MODIFIED LMDE COOLANT WATER COOLED	HEAT SINK	SEGMENTED WATER COOLED	NEW ABLATIVE MATERIALS	ADVANCED FLIGHT DESIGN																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																																											
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*FTP FIXED THRUST POSITION (9850 LBF)

** ALTITUDE TESTS

Figure 7-5. Test Matrix

2. MATERIALS SURVEY AND EVALUATION

As part of Program Phase II, an extensive survey of ablative and hard throat liquid propellant rocket engine combustion chamber materials was undertaken. Emphasis was placed upon development subsequent to 1962 to identify improved throat materials which could be incorporated into an advanced combustion chamber design for the Lunar Module Descent Engine. Detailed survey results are presented in Reference 21.

At the same time an ablative char reaction study was undertaken in the laboratory to determine which of a matrix of char specimens of different silica-to-carbon ratios best combined a high rate of endothermic reaction with a resulting interacted char structure of reasonably stable geometry. Results of the investigation are presented in References 16 and 20.

Finally, various ablative and hard throat candidate materials were evaluated in an inexpensive torch test that simulated LMDE combustion products at a representative heat flux, Reference 25.

2.1 MATERIALS SURVEY

The materials survey included a comprehensive review of the available literature on material developments supported by personal contacts with key government and industry groups active in the development, evaluation, and fabrication of rocket engine combustion chamber materials.

Insofar as ablative materials are concerned, only a few references were found that were applicable to the LMDE environment. Of those reports considered to be pertinent, none indicated a positive direction toward achievement of a significant improvement in throat erosion over that exhibited by the current LMDE silica fabric/phenolic laminate. There were indications that quartz fabric offers a potential improvement over silica fabric as a reinforcement. Therefore, the survey report (Reference 21) recommended that quartz fabric/phenolic be evaluated as a potential improvement in throat material for the LMDE as now constituted, despite the fact that quartz seemed to offer little advantage in an early LMDE throat test. Compared to injector performance at the time of the previous quartz/phenolic throat test, the current qualified injector provides more uniform propellant distribution and fuel barrier cooling so that the wall

environment is more uniform. This fact, plus the high viscosity of quartz compared to silica, could result in an overall reduction in throat erosion during the mission duty cycle.

On the basis of the survey of hard throat test results, it was concluded that several materials have the potential of reducing erosion as compared to ablative materials. However, no hard throat has yet shown the promise of zero erosion without cracking under simulated LMDE engine environment test conditions.

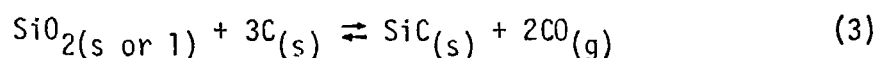
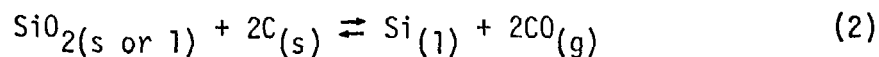
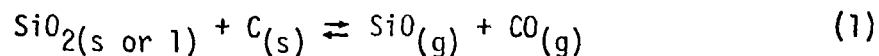
Considering the prime requirements for a hard throat of both chemical erosion resistance and thermal shock resistance, a detailed throat design analysis was recommended for four materials:

- (1) Tungsten/3-percent rhenium wire-reinforced zirconia
- (2) Si/SiC-infiltrated precharred graphite fabric laminate (Pyrocarb 751)
- (3) Hafnium-tantalum-coated tantalum-10 percent tungsten
- (4) JTA graphite.

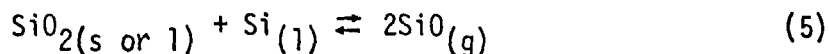
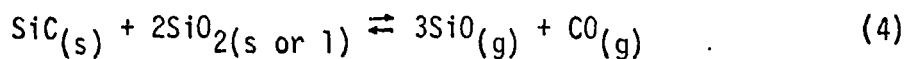
Items to be considered include the effect of variations of material thickness on insert thermal and stress gradients, in-wall temperatures, backwall temperatures, and surface operating temperatures, during the nominal LMDE mission duty cycle.

2.2 THE CARBON/SILICA REACTION

Highly endothermic silica-carbon reactions can take place at elevated temperatures within the char region of silica reinforced ablative materials. The primary char-reinforcement reactions which are thermodynamically feasible under typical LMDE chamber operating conditions are as follows:



Secondary reactions are also possible as follows:



It is not completely obvious, from the data presently available in the literature, whether the above reactions are beneficial or detrimental to the overall integrity of an ablative chamber in the LMDE combustion environment. On the one hand, all of the silica-carbon reactions absorb large quantities of heat. On the other hand, however, at least one of these reactions, $\text{SiO}_{2(s)} + \text{Si}_{(l)} \rightleftharpoons 2\text{SiO}_{(g)}$ may contribute markedly to the degradation of the geometrical integrity of the char structure.

Commercially-available silica- or quartz-reinforced ablative materials which are candidates for evaluation under Contract NAS9-8229 represent a wide range of silica-to-carbon char ratios and silica-carbon char microstructures. Hence, these materials can be expected to behave quite differently with respect to both the nature and rate of the endothermic char reactions. For example, the presently used silica-phenolic LMDE chamber material has a high silica-to-carbon ratio which should favor the ultimate formation of volatile silicon monoxide. On the other hand, material systems with lower silica-to-carbon ratios may well favor the formation of refractory silicon carbide (Reaction 3).

2.2.1 Test Specimens

Table 2-1 describes the matrix of test specimens which were evaluated in the TRW laboratory investigation (Reference 16). The specimens were prepared in the form of small cylinders (0.25 inch diameter by 0.25 inch long) by either laminating or molding techniques. The molded specimens contained chopped silica and/or carbon fibers which were randomly oriented throughout the resin matrix. In samples TRW-5 and TRW-6, a small amount of -325 mesh silicon carbide powder was added to the molding compound in addition to the silica and carbon chopped fibers. In samples TRW-7 and TRW-8, -325 mesh silicon powder was added. The laminated specimens were machined from flat blocks with the cloth plies oriented parallel to the center line of the cylinder. All of the specimens were charred prior to testing by heating in a vacuum at 1000°C for approximately 2 hours.

Sample Designation	Sample Type
MX-2600	Flat Laminate
TRW-1	Molded Cylinder
TRW-2	Molded Cylinder
TRW-3	Molded Cylinder
TRW-4	Molded Cylinder
TRW-5	Molded Cylinder
TRW-6	Molded Cylinder
TRW-7	Molded Cylinder
TRW-8	Molded Cylinder
TRW-9	Flat Laminate
TRW-10	Flat Laminate
TRW-11	Flat Laminate
TRW-12	Molded Cylinder

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Table 2-1. Matrix of Specimens for High Temperature Char Interaction Studies

Type	Resin Type and % by wt	Reinforcement Type and % by wt	Filler Type and % by wt	Resulting Initial Char Composition			
				% SiO ₂	% C	% SiC	% Si
minate	Phenolic SC1008 31%	Silica Cloth Type C100-48 61%	Silica Powder 8%	81%	19%	-	-
Cylinder	Phenolic SC1008 31%	Silica Yarn R10060 69%	-	81%	19%	-	-
Cyl	Phenolic SC1008 30%	Carbon Yarn Type CR10800 27% Silica Yarn Type R10060 43%	-	50%	50%	-	-
Cylinder	Phenolic SC1008 30%	Carbon Yarn Type CR10800 17% Silica Yarn Type R10060 53%	-	62%	38%	-	-
Cylinder	Phenolic SC1008 18%	Silica Yarn R10060 82%	-	90%	10%	-	-
Cylinder	Phenolic SC1008 30%	Silica Yarn R10060 46% Carbon Yarn Type CR10800 15%	Silicon Carbide Powder 9%	56%	34%	10%	-
Cylinder	Phenolic SC1008 20%	Silica Yarn Type R10060 67% Carbon Yarn Type CR10800 5%	Silicon Carbide Powder 8%	73%	17%	10%	-
Cylinder	Phenolic SC1008 30%	Silica Yarn Type R10060 45% Carbon Yarn Type CR10800 15%	Silicon Metal Powder 9%	56%	34%	-	10%
Cylinder	Phenolic SC1008 20%	Silica Yarn Type R10060 67% Carbon Yarn Type CR10800 5%	Silicon Metal Powder 8%	73%	17%	-	10%
minate	Phenolic SC1008 30%	Silica Cloth Type C100-48 48% Carbon Cloth Type CCA5 22% (Alternate Layers of Carbon and Silica)	-	57%	43%	-	-
minate	TRW A Type Polyimide -35%	Silica Cloth Type C100-48 65%	-	73%	27%	-	-
minate	TRW A Type Polyimide -35%	Silica Cloth Type C100-48 44% Carbon Cloth Type CCA5 21% (Alternate Layers of Carbon and Silica)	-	49%	51%	-	-
	Phenolic SC1008 55%	Silica Yarn Type R10060 45%	-	62%	38%	-	-

2.2.2 Experimental Apparatus and Testing Procedures

Figure 2-1 shows a schematic diagram of the high temperature, constant volume, reaction kinetics system used in all of the experiments. The system consisted of an Astro high temperature furnace, automatic temperature control system and recorder, Mullite reaction tube, gas manifold and storage bulbs, high-vacuum pumping system, a sensitive Pirani Gauge, and recording devices for continuous monitoring of the system pressure. Under operation, the volume of the system was constant, and the rate of reaction was followed by the pressure rise in the system as carbon monoxide gas was evolved from the reacting silica-carbon pellets.

Mass spectroscopic examination of the gas generated from typical experiments at 1500°C showed that carbon monoxide was the predominant (>90 percent) non-condensable gaseous reaction product. As indicated previously, silica can react with carbon in several ways to produce carbon monoxide as a reaction product.

All of the reactions were thermodynamically feasible under the low pressure (1×10^{-3} to 3 torr) experimental conditions. Evidence for determining which of the preceding reaction schemes predominated was obtained by measurement of initial and final sample weights for each specimen, total carbon monoxide generated, and by X-ray diffraction analysis of the resulting solid products of reaction.

2.2.3 Test Results

Table 2-2 and Figures 2-2 and 2-3 summarize the experimental data.

The first column of Table 2-2 presents the sample designation for each of the 13 materials evaluated. Composition and fabrication approach for each of the 13 materials was presented previously in Table 2-1.

The second column of Table 2-2 lists the initial silica/carbon weight ratio of each of the charred specimens. It is apparent from this column that the test specimens cover a wide range of initial silica-to-carbon ratios. The MX-2600 reference material ($\text{SiO}_2/\text{C} = 4.3$) is near to stoichiometric for the formation of silicon monoxide (reaction 1), while specimens TRW-3, TRW-5, TRW-7, and TRW-12 result in SiO_2/C char ratios which are just stoichiometric for the formation of silicon carbide (reaction 3).

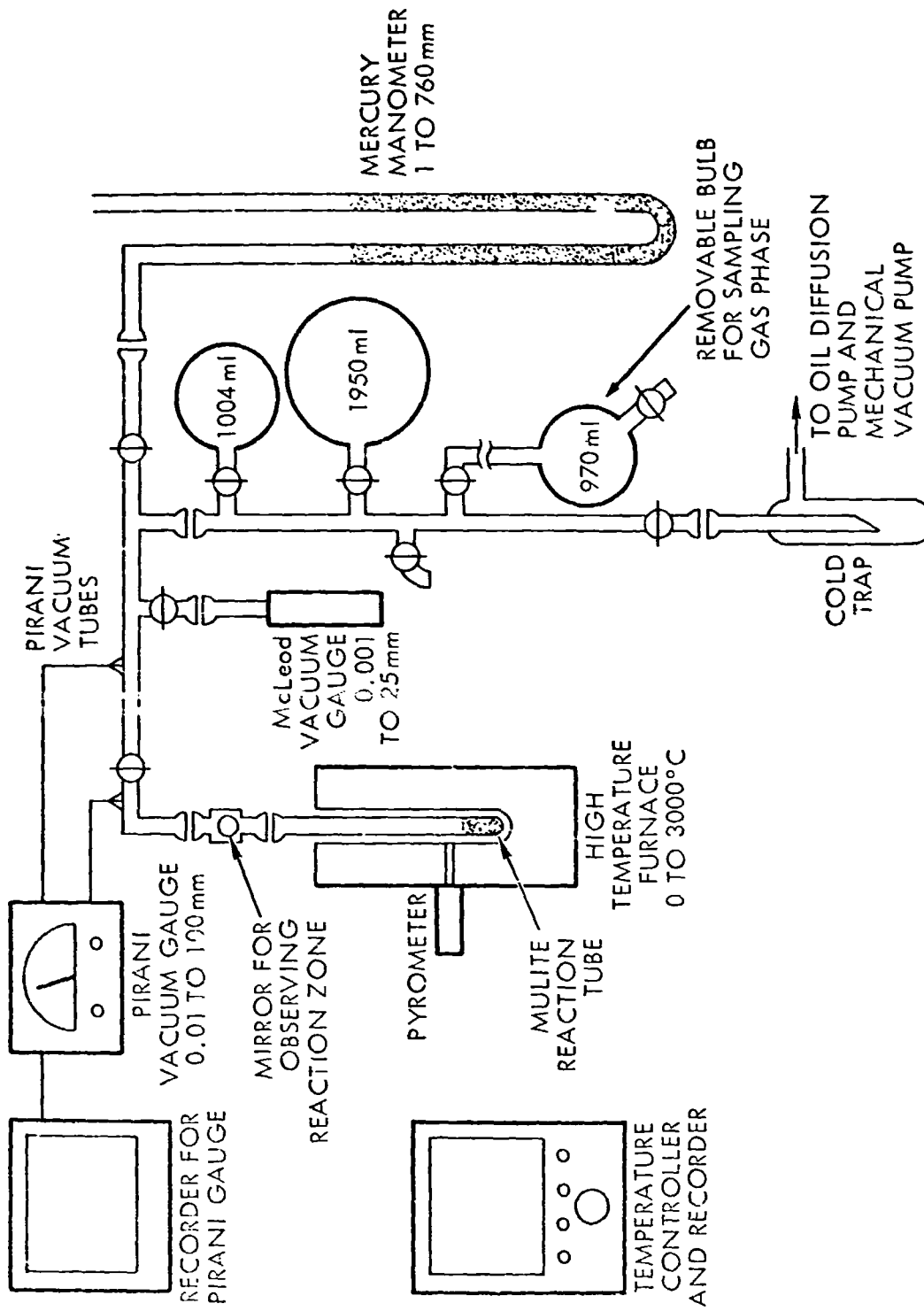


Figure 2-1. Schematic Diagram of the Constant Volume Reactor System

Table 2-2. Summary of Experimental

Sample Designation	Initial SiO ₂ /C Ratio of Char	Measured Weight Loss* mg	Dimensional Change of the Sample	X-ray Count of the Sample
MX-2600	4.3	124	Reduced to Powder	
TRW-1	4.3	120	Reduced to Powder	
TRW-2	1.0	76	No Change	α SiC Crystal
TRW-3	1.6	87	No Change	α SiC Crystal
TRW-4	9.0***	73	32% Diameter Shrinkage	α SiO ₂ -SiC
TRW-5	1.6	81	No Change	α SiC Crystal
TRW-6	4.3	100	Reduced to Powder	
TRW-7	1.6	77	No Change	α SiC Crystal
TRW-8	4.3	112	5% Diameter Shrinkage	
TRW-9	1.3	70	Slight - No Bond Between Laminations	
TRW-10	2.7	112	Powder	
TRW-11	1.0	71	Slight - No Bond Between Laminations	
TRW-12	1.6	94	4% Diameter Shrinkage	α SiC Crystal

*The reported weight losses include the specimen weight to the 1500°C environment.

**Calculated weight changes were based on the assumption from the reaction tube and deposit on the cooler rod. In Samples TRW-5 through TRW-8, the relatively small calculations.

***More CO formed in this experiment than could be accounted for. This specimen was higher than the 10% reported.

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Measurements and Comparison of Measured and Calculated Sample Weight Changes

Diffraction tification on-Volatile tion Products	Calculated Weight Loss** - mg			Most Probable Reactions
	Assuming SiO Forms (reaction 1)	Assuming Si Forms (reaction 2)	Assuming SiC Forms (reaction 3)	
	135	86	64	Reaction 1
	142	91	67	Reaction 1
Only Detected line Specie	124	80	60	Reaction 2 or Com- bination of 1 and 3
Only Detected line Specie	142	91	67	Reaction 2 or Com- bination of 1 and 3
2 Major Phase ot Detected	103	65	47	Reaction 1***
Only Detected line Specie	127	82	61	Reaction 2 or Com- bination of 1 and 3
	102	65	47	Reaction 1
Only Detected line Specie	121	79	59	Reaction 2 or Com- bination of 1 and 3
	94	60	44	Reaction 1
	81	61	47	-
	122	81	61	Reaction 1
	82.5	57.5	45.5	Peaction 1
Only Detected line Specie	235	151	112	Reaction 3

loss during pre-charring as well as the weight loss during 30-minute exposure of the char

that all SiC which forms remains in the sample tube, and all SiO or Si which forms evaporate from the system. Calculated weight changes were based on measured CO pressure rise data. Amounts of Si or SiC initially present were not taken into account in making the weight loss

ated for by any of the three reactions. It would appear likely that the actual carbon content

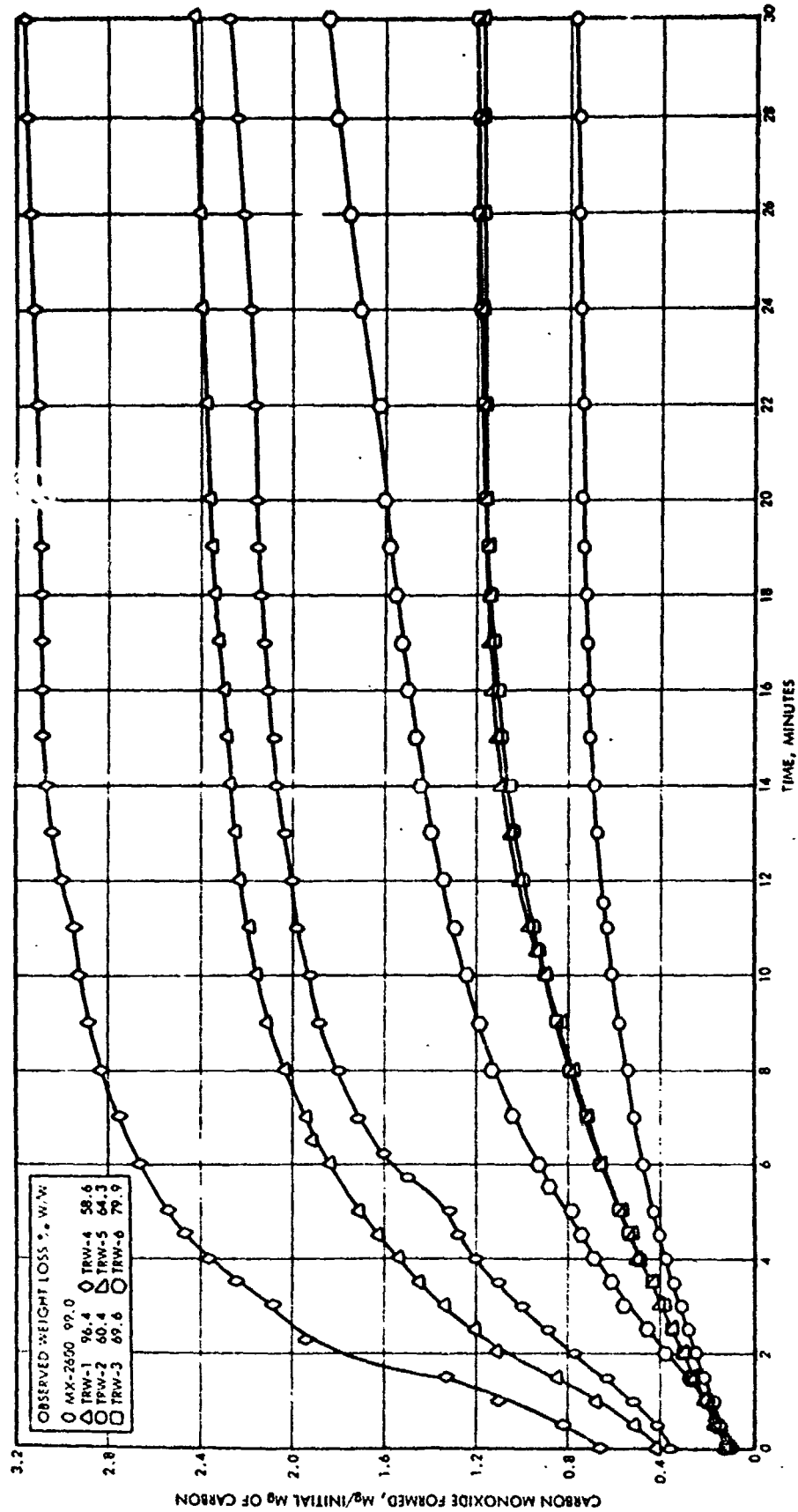


Figure 2-2. Carbon Monoxide Formation as a Function of Time for TRW Specimens 1 through 6 and MX-2600

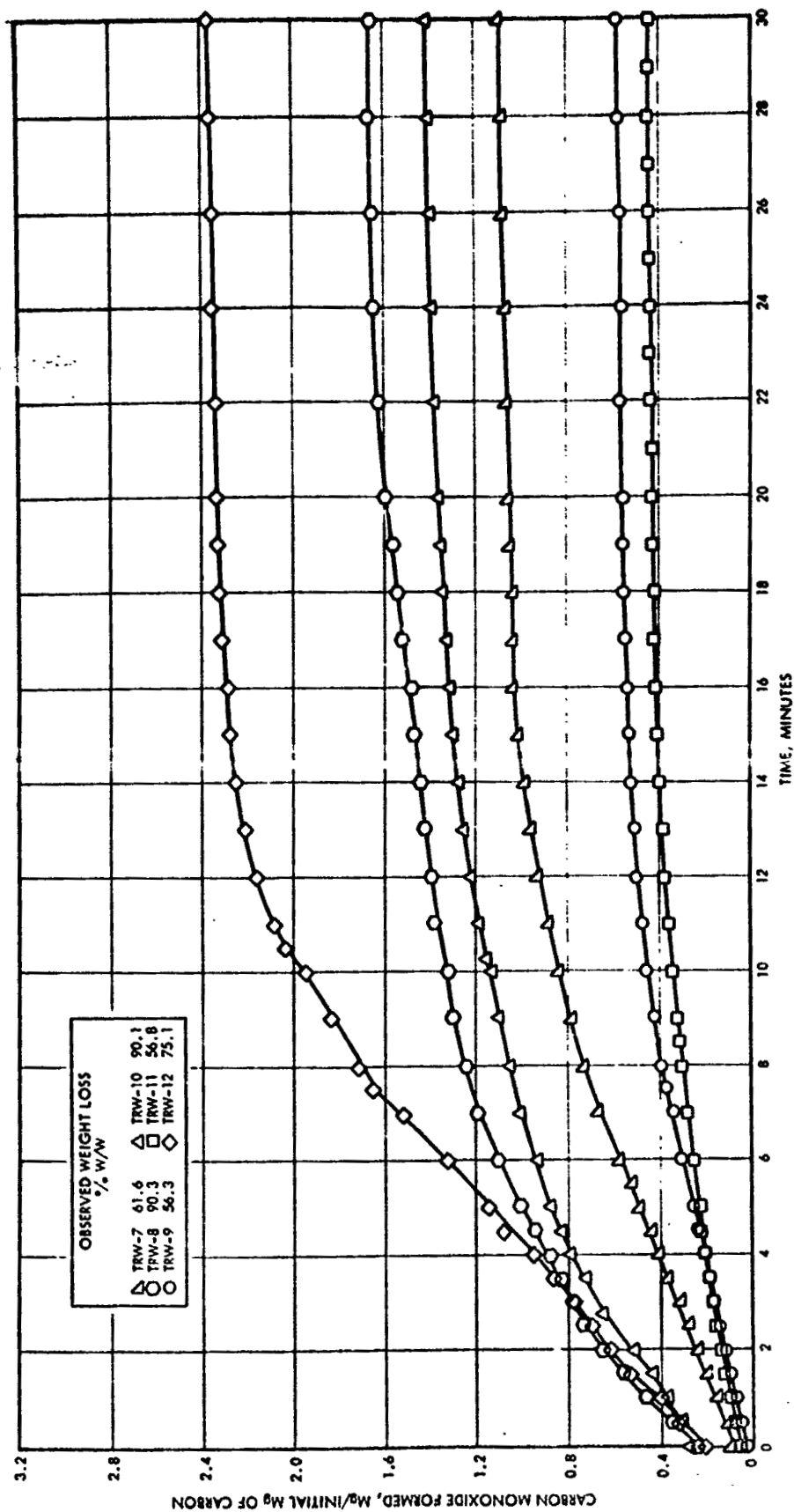


Figure 2-3. Carbon Monoxide Formation as a Function of Time for TRW Specimens 7 through 12

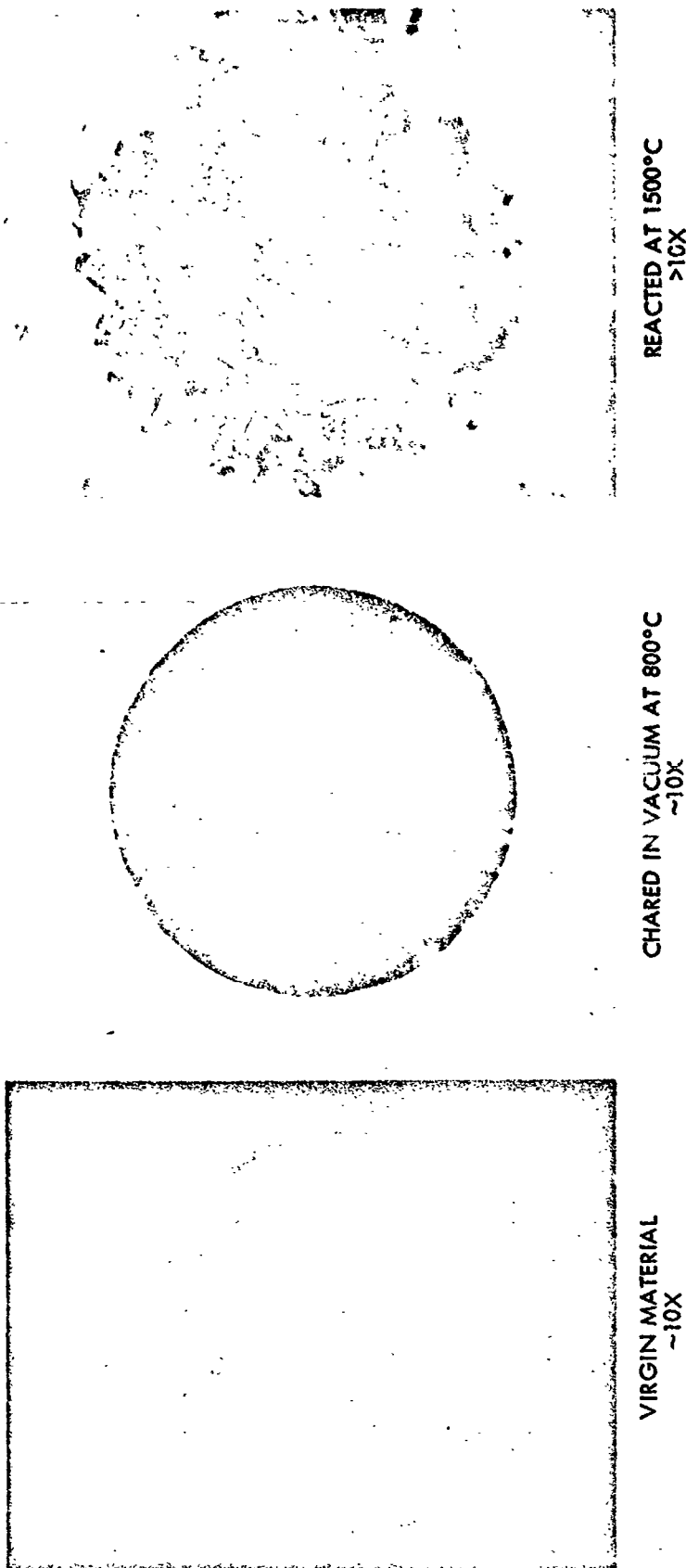
The third column of the table lists the measured weight loss of each specimen in milligrams. The reported weight losses include the specimen weight loss during pre-charring as well as the weight loss during 30-minute exposure of the char to the 1500°C temperature. Initial sample weights were 125 ± 1 mg in all cases. From a comparison of columns two and three, it is apparent that the samples with high initial silica-to-carbon ratios had the greater weight losses.* Indeed, the MX-2600 reference material and the TRW-1 sample both reacted to the point where greater than 95% of the initial sample was converted to volatile products after the 30-minute, 1500°C test.

The fourth column of Table 2-2 indicates the magnitude of dimensional changes in the specimens which occurred during the 30-minute exposure to the 1500°C test temperature. It is apparent from this column that there were great differences in the abilities of the different candidate material specimens to maintain their geometrical integrity during exposure to testing conditions. These differences are illustrated in Figures 2-4 through 2-16, which show photomicrographs of the specimens before and after testing. Nearly all specimens with silica-to-carbon ratios near to stoichiometric for the formation of silicon monoxide (MX-2600 and TRW No. 1, 6, and 10) were reduced to powder during the test, while many of the specimens with low silica-to-carbon ratios (TRW No. 2, 3, 5, and 7) exhibited either no change or a very small change in dimensions during the thermal exposure. The differences in geometrical integrity and sample weight loss between the different types of candidate materials strongly suggests that a different reaction mechanism predominated for specimens with high silica-to-carbon ratios than predominated for low silica-to-carbon ratio materials. Possible exceptions to this conclusion were specimens TRW-9 and TRW-11 which were prepared by laminating alternate layers of silica and carbon cloth. The photomicrographs of these specimens indicate that reaction took place predominantly in the silica-rich areas of the specimens, and even though the overall silica-to-carbon ratios of these specimens was

* Sample TRW-4 is an exception which is readily explained. This specimen had such a large excess of silica over that which could theoretically react that the total weight loss could not be very high.

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MX - 2600 (CHAR = 81% SiO_2 , 19% C)

Figure 2-4



VIRGIN MATERIAL
~10X



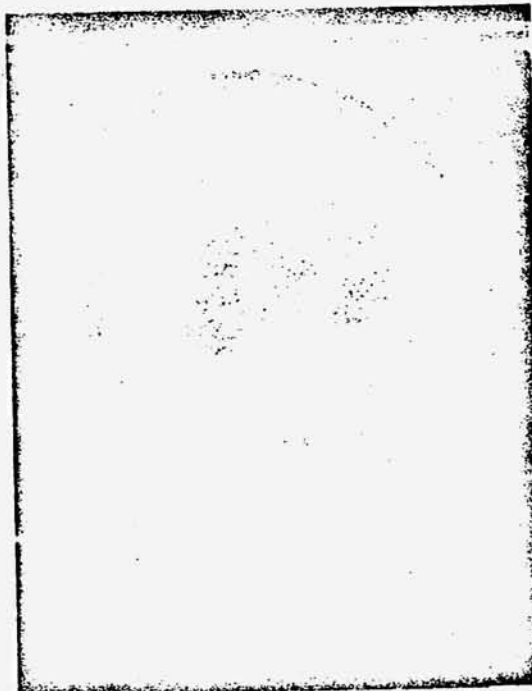
AFTER 1500°C REACTION
>10X

TRW - 1 (CHAR = 81% SiO_2 , 19% C)

Figure 2-5



VIRGIN MATERIAL
~10X



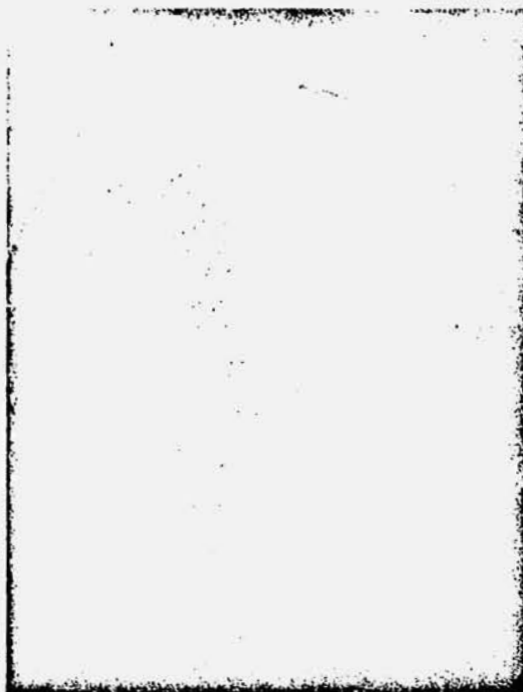
AFTER 1500°C REACTION
~10X

TRW - 2 (CHAR = 50% SiO_2 , 50% C)

Figure 2-6



VIRGIN MATERIAL
~10X



AFTER 1500°C REACTION
~10X

TRW - 3 (CHAR = 62% SiO_2 , 38% C)

Figure 2-7



VIRGIN MATERIAL
~10X



AFTER 1500°C REACTION
~10X

TRW - 4 (CHAR = 90% SiO_2 , 10% C)

Figure 2-8



VIRGIN MATERIAL
~10X



AFTER 1500°C REACTION
~10X

Figure 2-9

TRW - 5 (CHAR = 56% SiO_2 , 34% C, 10% SiC)



VIRGIN MATERIAL
~10X



AFTER 1500°C REACTION
~10X

Figure 2-10

TRW - 6 (CHAR = 73% SiO_2 , 17% C, 10% SiC)



VIRGIN MATERIAL
~10X



REACTED AT 1500°C
~10X

Figure 2-11

TRW - 7 (CHAR = 56% SiO_2 , 34% C, 10% Si)



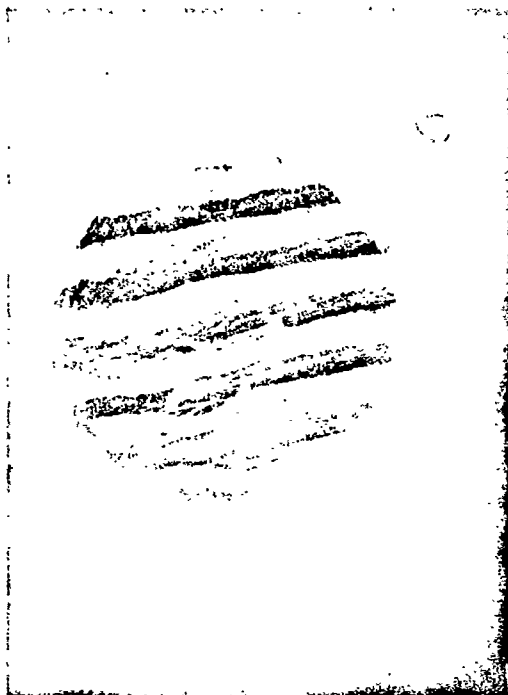
VIRGIN MATERIAL
~10X



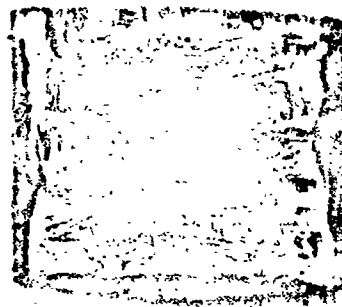
REACTED AT 1500°C
~10X

Figure 2-12

TRW - 8 (CHAR = 73% SiO_2 , 17% C, 10% Si)

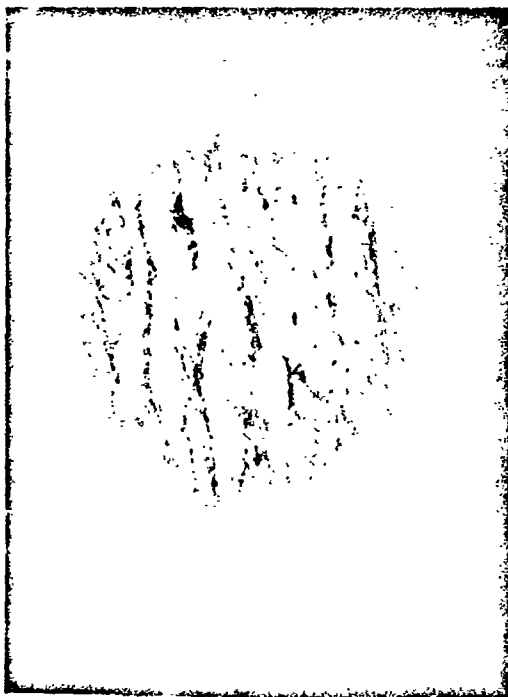


VIRGIN MATERIAL
~10X

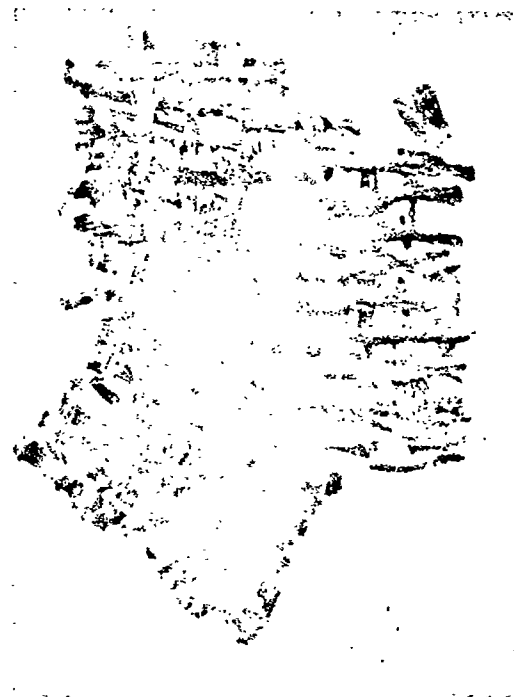


REACTED AT 1500°C
CROSS SECTION ~10X

Figure 2-13
TRW - 9 (CHAR = 57% SiO_2 , 43% C)



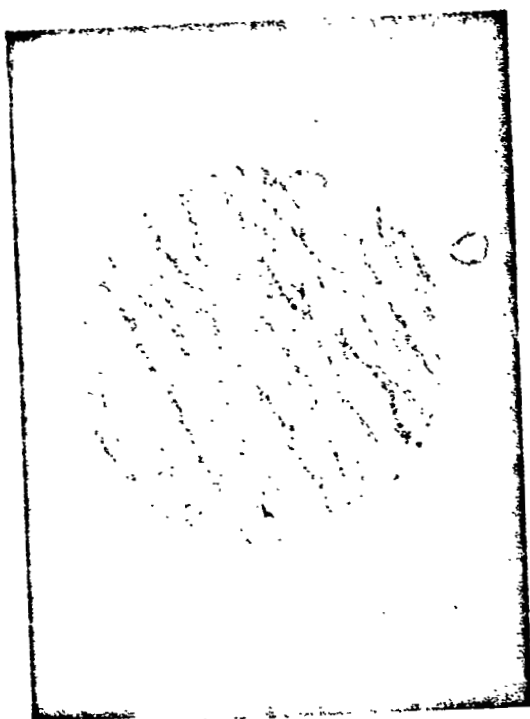
VIRGIN MATERIAL
~10X



REACTED AT 1500°C
>10X

Figure 2-14
TRW - 10 (CHAR = 73% SiO_2 , 27% C)

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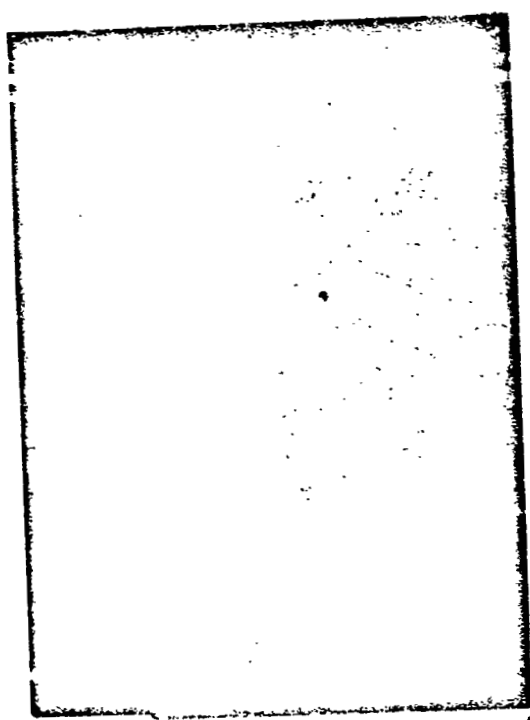


VIRGIN MATERIAL
~10X

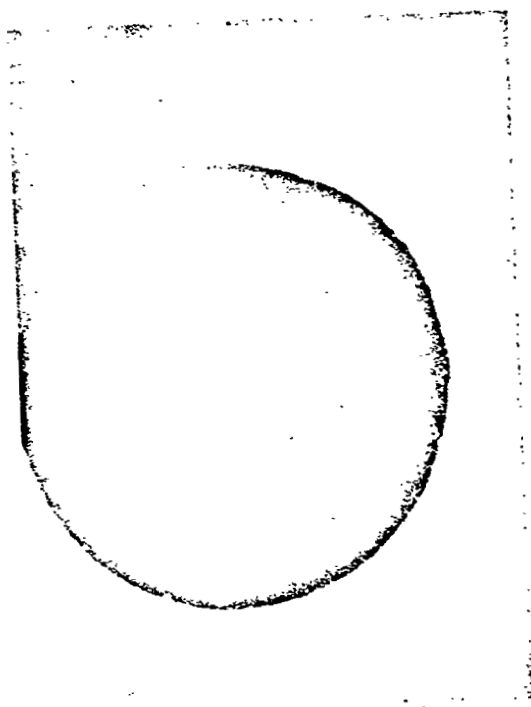


REACTED AT 1500°C
CROSS SECTION ~10X

Figure 2-15
TRW - 11 (CHAR = 49% SiO₂, 51% C)



VIRGIN MATERIAL
~10X



REACTED AT 1500°C
~10X

Figure 2-16
TRW - 12 (CHAR = 62% SiO₂, 38% C)

very low, it appears likely that the localized silica-rich areas of the specimens resulted in a predominant reaction mechanism similar to what occurred in specimens of high overall silica-to-carbon ratio.

The fifth column describes the results of X-ray diffraction analyses of selected specimens after reaction. Analysis of the specimens with low silica-to-carbon ratios (TRW No. 2, 3, 5, and 7) indicated that alpha silicon carbide was the only detectable crystalline species present. It is possible, of course, that unreacted amorphous carbon was also present in some or all of these specimens to a significant extent, since amorphous materials cannot be detected by the X-ray diffraction analysis. Specimen TRW-4 which had a very high silica-to-carbon ratio showed a very different result from the low silica-to-carbon ratio specimens. No silicon carbide was detected in sample TRW-4, and α silica (SiO_2) was the predominant species present. Small amounts of silicon and graphite may also have been present in this specimen.

The next three columns of Table 2-2 show calculated sample weight losses based on measured carbon monoxide pressure rise data assuming that either reaction 1 (SiO forms), reaction 2 (Si forms), or reaction 3 (SiC forms) takes place. The calculated weight changes were based on the assumption that all SiC which forms remains in the sample tube, and all SiO and Si which form evaporate from the reaction tube and deposit in the cooler regions of the system. In samples TRW-5 through TRW-8, the relatively small amounts of Si or SiC initially present were not taken into account in making the weight loss calculations. A comparison of the measured weight loss data presented in column 3 with the calculated weight changes shown in columns 6 through 8 shows that weight changes calculated on the basis of reaction 1 (SiO formation) give the best correlation with the measured weight changes for all of the specimens with high initial silica-to-carbon ratios (samples HX-2600, TRW-1, TRW-4, TRW-6, TRW-8, and TRW-10). On the other hand, specimens with lower silica-to-carbon ratios gave better correlations between measured and calculated weight changes when reaction 2, reaction 3, or a combination of reactions 1 and 3 were assumed to predominate. Since no silicon metal was detected in any of the

X-ray diffraction measurements on low silica-to-carbon specimens, it appears that a combination of reactions 1 and 3 is more likely for these samples than reaction 2.

The last column of Table 2-2 indicates the most probable reactions based on a comparison of the measured and calculated weight changes and the X-ray diffraction data. As indicated above, a reaction scheme leading to the formation of volatile silicon monoxide appears to predominate for the specimens with high silica-to-carbon ratio. On the other hand, for specimens with low initial silica-to-carbon ratios, reaction schemes leading at least in part to the formation of α silicon carbide appears to be favored.

Figures 2-2 and 2-3 show the reaction rate data for each of the samples plotted in terms of mg of CO formed per initial mg of carbon versus time. The initial reaction rates for the specimens in terms of CO formation varied over about a factor of 5. With the exception of sample TRW-12, the specimens with higher silica-to-carbon ratios appeared to have the higher reaction rates.

Experiments with samples TRW-2 and TRW-9 were rerun with some modifications to the basic test procedures described above. Instead of pre-charring the samples at 1000°C prior to 1500°C exposure, the samples were quickly heated from room temperature to 1500°C, held for 30 minutes at 1500°C, held for 30 minutes at 1500°C under vacuum, and then quickly cooled to room temperature. The test specimens after 1500°C exposure (no pre-charring) looked identical to the specimen in the initial experiments (pre-charred at 1000°C).

2.2.4 Laboratory Program Recommendations

Based on results of the TRW Laboratory studies that char reactions favoring the formation of silicon carbide, rather than volatile silicon monoxide, can be obtained by decreasing the initial silica-to-carbon ratio of the char below that of the current LMDE material, it was recommended that at least one ablative material containing a low silica-to-carbon char ratio be selected for engine testing. This ablative throat could be fabricated from standard phenolic resin and alternate layers of quartz fabric and graphite fabric. This composite system would yield a char with

a silica-to-carbon ratio of approximately 1.3:1, compared to a ratio of approximately 4.3:1 for the current MX-2600 material. A potential disadvantage of this approach is that the resultant char structure would be composed of silica-rich and carbon-rich areas which may be too far removed from each other to promote rapid chemical reaction.

Quartz fabric was proposed over silica because of its higher viscosity. Graphite fabric was proposed over carbon because of its higher temperature capability.

It was also recommended that another approach to enhancement of SiC formation be evaluated in a throat configuration. A high resin-content quartz-fabric/phenolic would be pre-charred and reimpregnated to its original silica-to-carbon ratio. This approach would have two advantages:

- (1) It would simplify part fabrication in that the need for interleaving dissimilar fabrics would be eliminated.
- (2) It would provide more intimate contact between the silica and carbon to enhance the reaction between the two.

2.3 TORCH TESTS

Many laboratory test methods have been used for the evaluation of chamber throat materials, but none can simulate exactly the conditions of the engine environments. The actual environment consists of at least four major components which combine, affecting materials behavior: thermal, reactive gas, velocity, and pressure. In the torch test investigation, reported completely in Reference 25, the methane-oxygen-nitric oxide mixture was used to approximate the thermal and reactive gas environment, but not the pressure and velocity effects. The purpose of the study was to evaluate and compare a large number of candidate throat materials for their thermo-chemical behavior under LMDE gas environments. Post torch test analysis criteria included visual examination, weight change, and dimensional change (erosion).

2.3.1 Test Equipment

The torch testing facility is pictured in Figure 2-17. The containment vessel is a stainless steel tank 4-feet in diameter by 7-feet high. The tank is divided at the center and flanged with an O-ring seal for vacuum-

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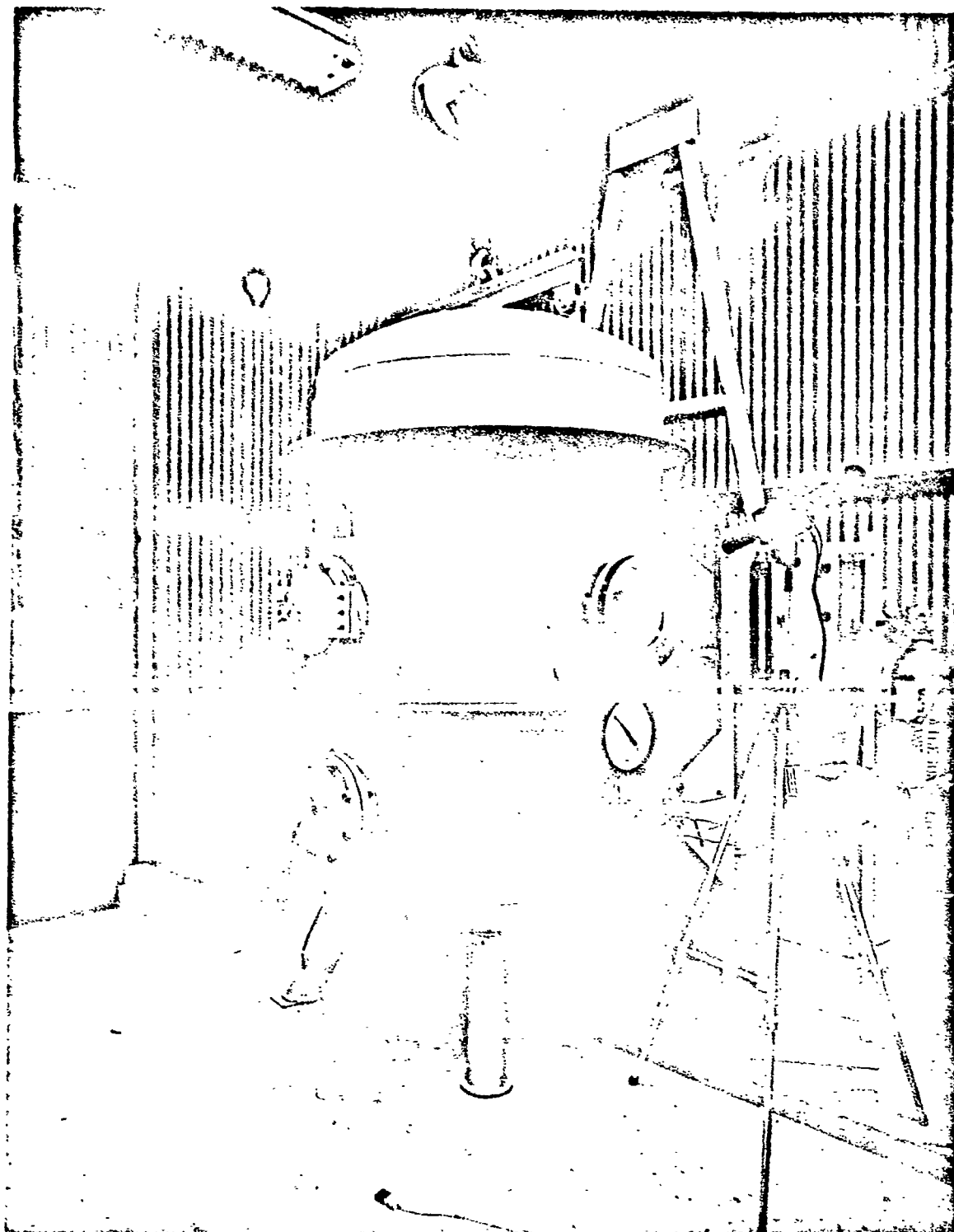


Figure 2-17. Torch Test Facility

tight operation. The upper chamber half is provided with three view ports for monitoring flame interaction experiments. Provisions are made to heat each view port to prevent fogging. The lower chamber half is equipped with two access ports through which metered gas flows are fed to the torch assembly and all electrical and instrumentation connections are made. The vacuum chamber is connected through a filtration system to a large roughing pump which exhausts to the atmosphere outside.

An A-frame and chain hoist are provided to facilitate removal of the upper chamber half which can be moved laterally to clear its lower section. All torch and sample handling controls and instrumentation are built into the lower half of the vacuum chamber. Test specimens to be exposed to the flame environment are supported on a circular platform. This plate can be positioned up or down to adjust torch-to-sample spacing. The plate pivots about the rear support to move experiments into and out of the established flame environment by means of an externally operated reach rod.

A schematic of the test apparatus is shown in Figure 2-18, and a closeup view of the torch test in progress is shown in Figure 2-19. The oxygen and nitric oxide gases are premixed prior to entry to the torch tip manifold where the oxidizers are mixed with methane before burning. The flow rates of the gases to the torch are metered with accurately calibrated flow-meters. A two-color pyrometer is positioned outside the chamber to monitor the surface temperature of the test specimen during testing.

2.3.2 Test Procedure

Test conditions were established to maximize the thermal environments and to approximate an oxygen-rich LMDE chamber gas composition. A computer program based on thermodynamic data of $\text{CH}_4\text{-O}_2\text{-NO}$ was used to guide the selection of the desired gas mixture. After experimenting with a number of fuel/oxidizer ratios using MX-2600 specimens and comparing with the computer program output on equilibrium chemical species produced from these mixtures, a composition of 40 CH_4 -50 O_2 -10 NO by volume was selected as the best compromise of the flame temperature and combustion products. This flame was used in all torch testing in this program unless otherwise specified.

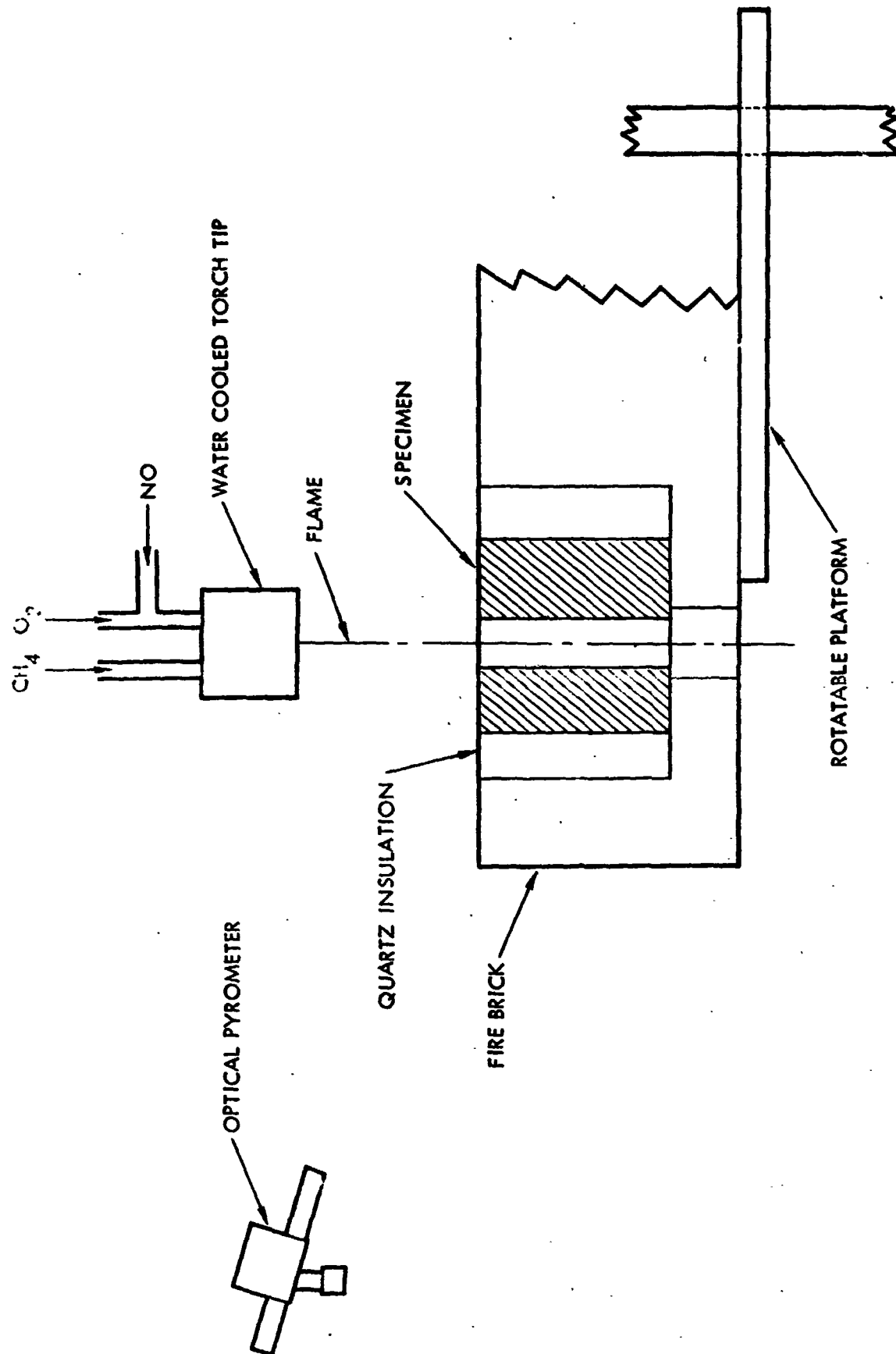


Figure 2-18. Schematic of Test Apparatus

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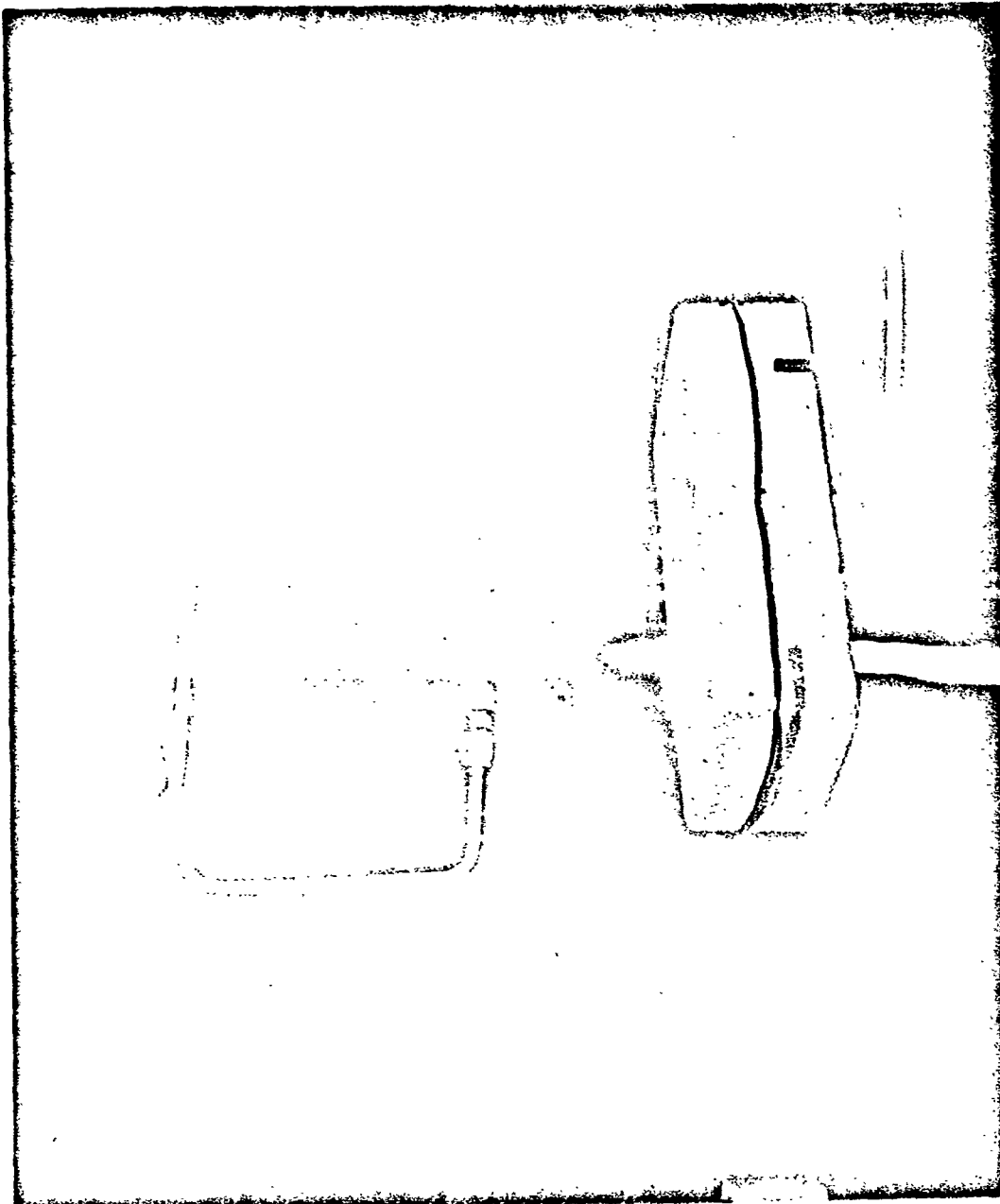


Figure 2-19. Torch Test in Progress

A comparison of equilibrium torch temperature and chemical species with LMDE chamber gas composition and temperature for a number of mixture ratios is shown in Table 2-3.

The heat flux was determined with a copper calorimeter having the same configuration as the test specimen, Figure 2-20. The heat flux radially through the wall of the specimen was found to be 120 Btu/sq. ft/sec and was checked periodically to assure uniformity of test condition from specimen to specimen.

Having standardized the gas mixture and heat flux, the next step was to determine the most appropriate test duration for comparing the various materials under the same test conditions. Experiments were conducted with MX-2600 for test durations of 150 seconds, 300 seconds, 600 seconds, 1200 seconds and 1800 seconds. Results of weight losses and diametrical changes are tabulated in Table 2-4 and plotted in Figure 2-21.

On the basis of the MX-2600 test results, 300-second and 600-second test durations were selected as reference points for evaluation and comparison of candidate throat materials. Therefore, all testing in this investigation was performed using the two time durations, unless otherwise specified. Test specimen preparation and manufacturing methods for these materials are described in detail in Reference 25. In the main text of this report, the commercial names or brief descriptions of the materials are used rather than the part numbers of Reference 25.

The weight and dimensions of each specimen were measured before and after torch testing. In the beginning of the run, the heat flux was measured and the flame and/or specimen positions were adjusted to assure alignment and concentricity of the flame with the center line of the test specimen. After evacuating the chamber and with the test specimen in place on the platform, but out of the test position, the flame was lit using the desired gas mixture. The specimen was then swung into the test position as timing started. Temperature readings were taken at 30-second intervals by focusing the pyrometer on the inside diameter surface about 1/16-inch below the top edge of the hole. At the conclusion of the test duration, gas flow was immediately shut off and the specimen was allowed to cool naturally while the vacuum was maintained.

Table 2-3. Chemical Species in Oxygen/Fuel Mixtures

Mixture	Condition	Temperature °R	Mole Fraction of Chemical Species at Equilibrium					
			CO	CO ₂	H ₂	H ₂ O	N ₂	O ₂
N ₂ O ₄ /50-50 N ₂ H ₄ -UDMH MR = 1.6 P _C = 100 psi	LMDE Throat	5180	9.4 x 10 ⁻²	3.7 x 10 ⁻²	1.3 x 10 ⁻¹	3.6 x 10 ⁻¹	3.3 x 10 ⁻¹	1.7 x 10 ⁻³
Same as Above	LMDE Chamber	5502	3.5 x 10 ⁻²	3.4 x 10 ⁻²	1.3 x 10 ⁻¹	3.5 x 10 ⁻¹	3.3 x 10 ⁻¹	3.0 x 10 ⁻³
40 CH ₄ - 50 O ₂ - 10 NO by Volume	Torch Test	5407	2.3 x 10 ⁻¹	6.9 x 10 ⁻²	1.7 x 10 ⁻¹	3.7 x 10 ⁻¹	3.6 x 10 ⁻²	1.0 x 10 ⁻²
50 CH ₄ - 50 O ₂ by Volume	Torch Test	4707	2.5 x 10 ⁻¹	3.7 x 10 ⁻²	3.5 x 10 ⁻¹	2.8 x 10 ⁻¹	-0-	7.8 x 10 ⁻⁵

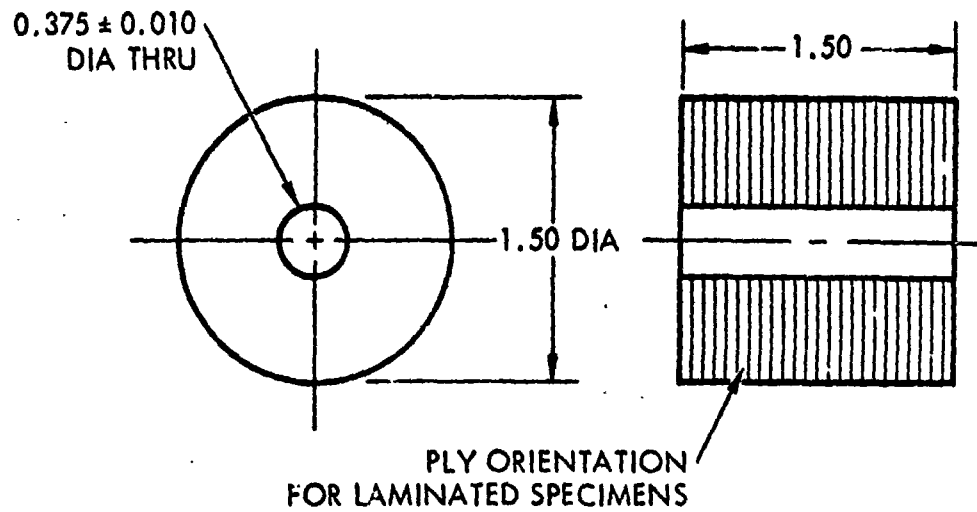


Figure 2-20. Torch Test Specimen Configuration

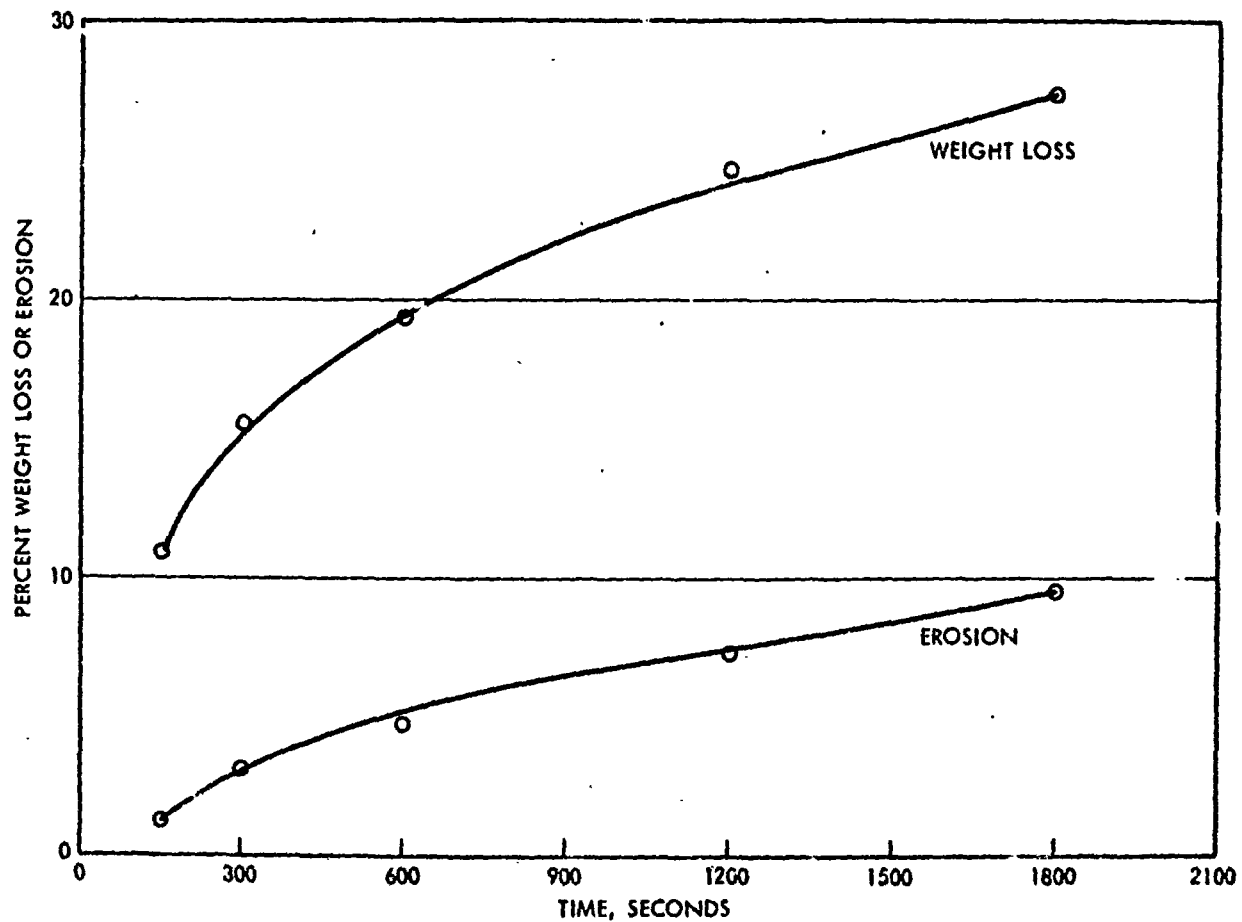


Figure 2-21. Rates of Weight Loss and Erosion of MX-2600

Table 2-4. Weight Loss and Percent Erosion of MX-2600 from Torch Testing

Test Duration, Seconds	Weight Loss, %	Erosion, %
150	10.8	1.3
300	15.5	3.1
600	19.4	4.8
1200	24.7	7.4
1800	27.3	9.6

2.3.3 Test Results

Post-test analyses included visual examination, weight loss determination, and erosion or dimensional changes. Metallographic and X-ray diffraction analyses were performed on certain specimens as required. Photographs of torch-test specimens are shown by groups in Figures 2-22 through 2-25. Table 2-5 summarizes the complete test data showing the equilibrium surface temperature, test duration, weight loss and percent erosion. Results are discussed in Reference 25.

Typical specimen close-ups are shown in Figures 2-26 and 2-27. Hard throat material erosion along the length of the test specimen is compared to that of MX-2600 silica-fabric/phenolic in Figure 2-28. The erosion profile of a typical quartz-fabric/phenolic, MXQ-190, is compared in Figures 2-29 and 2-30. The erosion profile of a typical pyrolyzed laminate, Carbitex 100, is shown in Figure 2-31.

2.3.3.1 Gas Composition Effect

All torch testing had been conducted with an oxidizer-rich flame simulating the most severe LMDE chamber environment. An apparent anomaly was the fact that there was no trace of silicon carbide in any of the specimens exposed to the torch, even those formulations that did form silicon carbide in laboratory experience. In addition, silicon carbide was also found in the surface char of LMDE chamber liners that had been exposed to a duty-cycle firing. It was postulated, therefore, that at full thrust operation, the boundary layer of the LMDE chamber was fuel-rich and that fact promoted the in situ reaction between carbon and silica

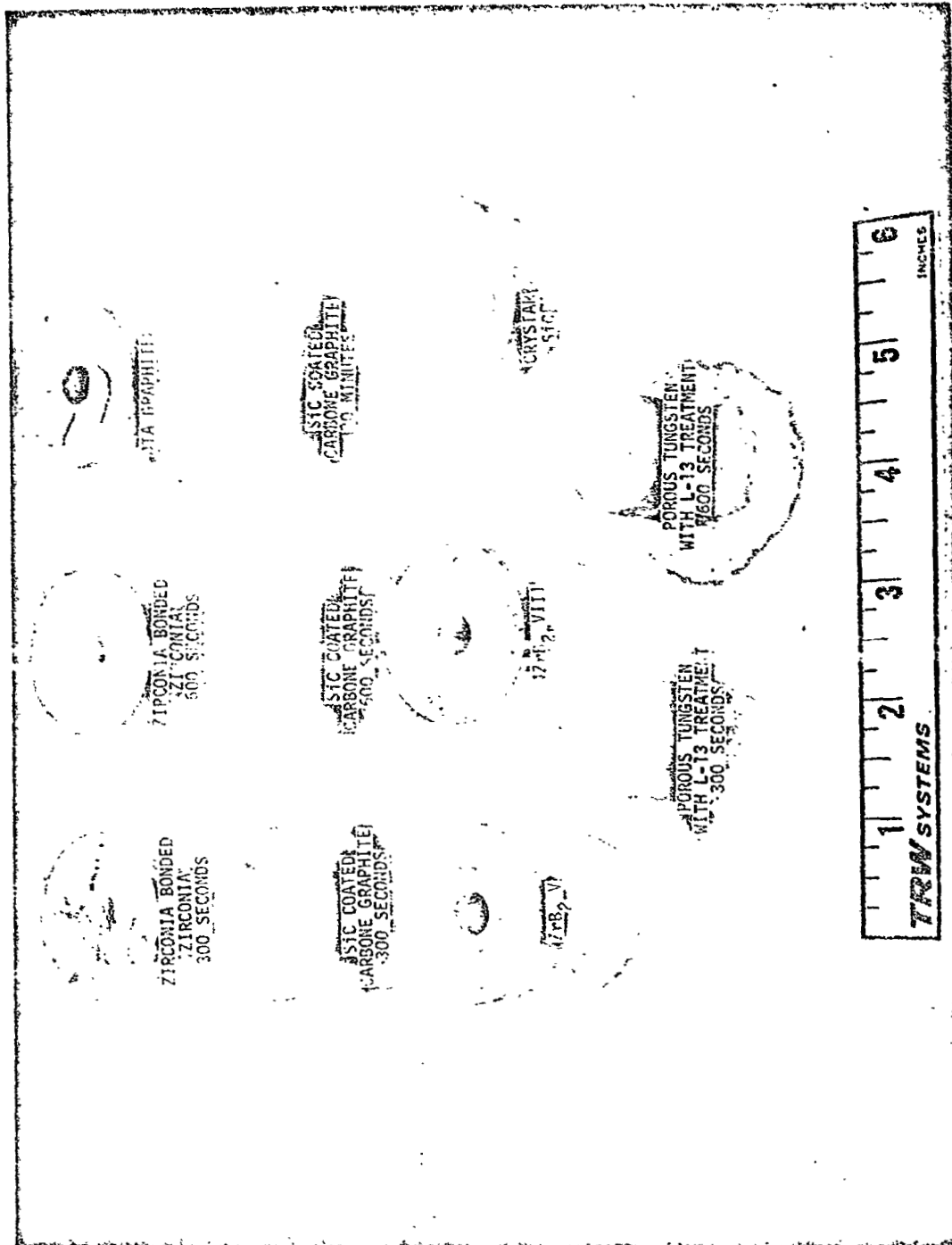


Figure 2-22 "Hard Throat" Materials after Torch Test

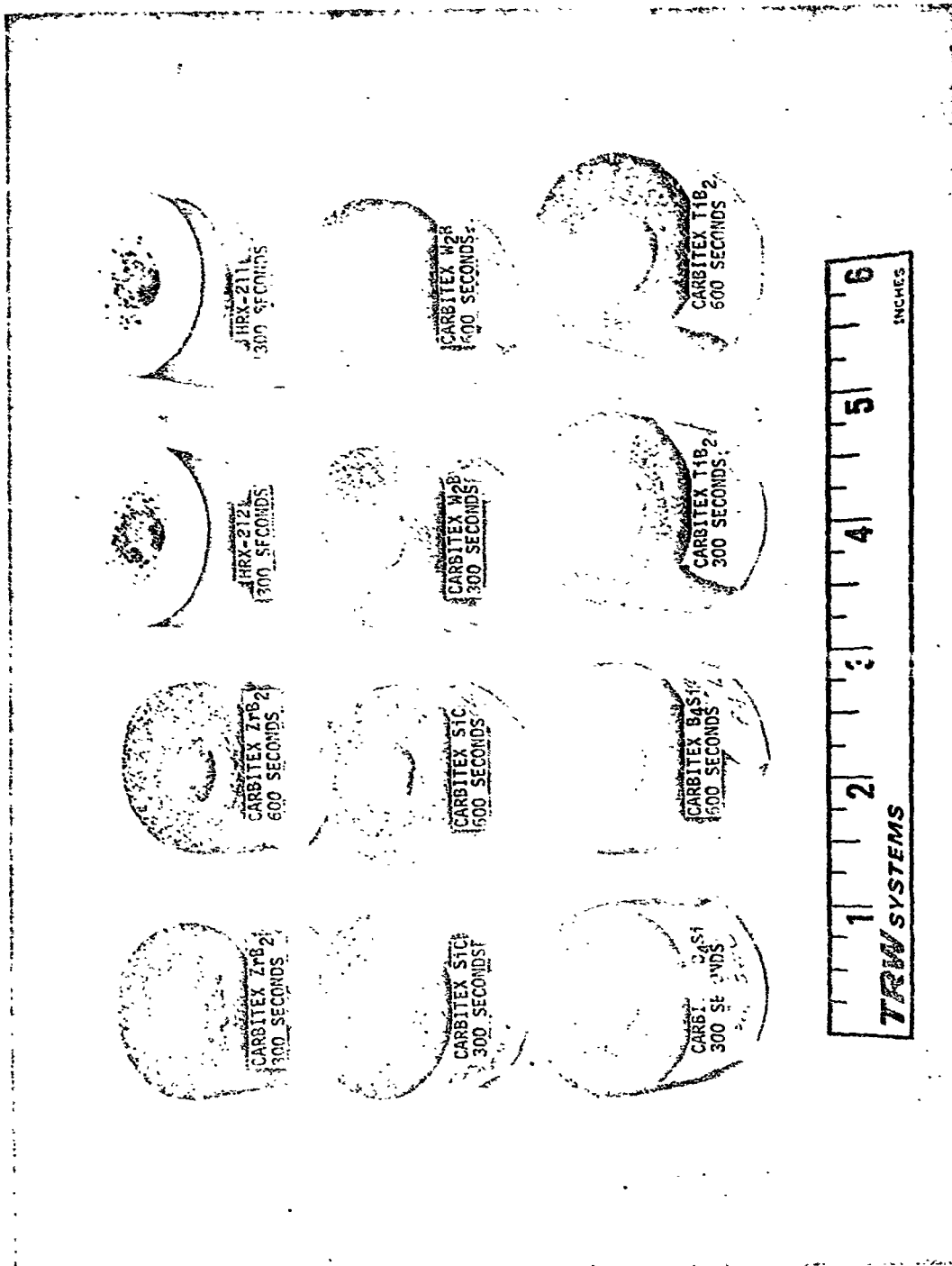


Figure 2-23 Pyrolyzed Laminate Materials after Torch Test

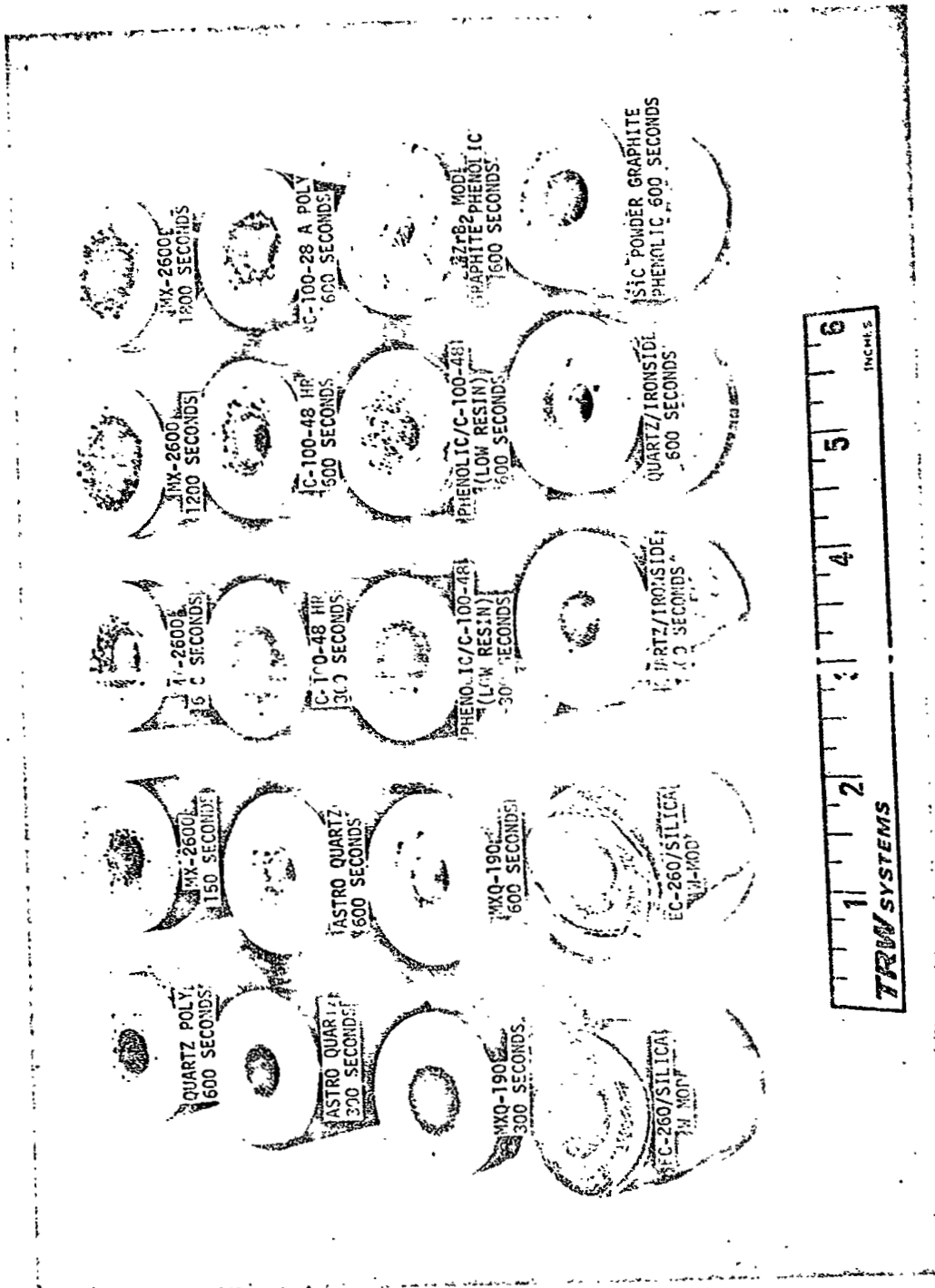


Figure 2-24 Resin-Base Laminate Materials after Torch Test

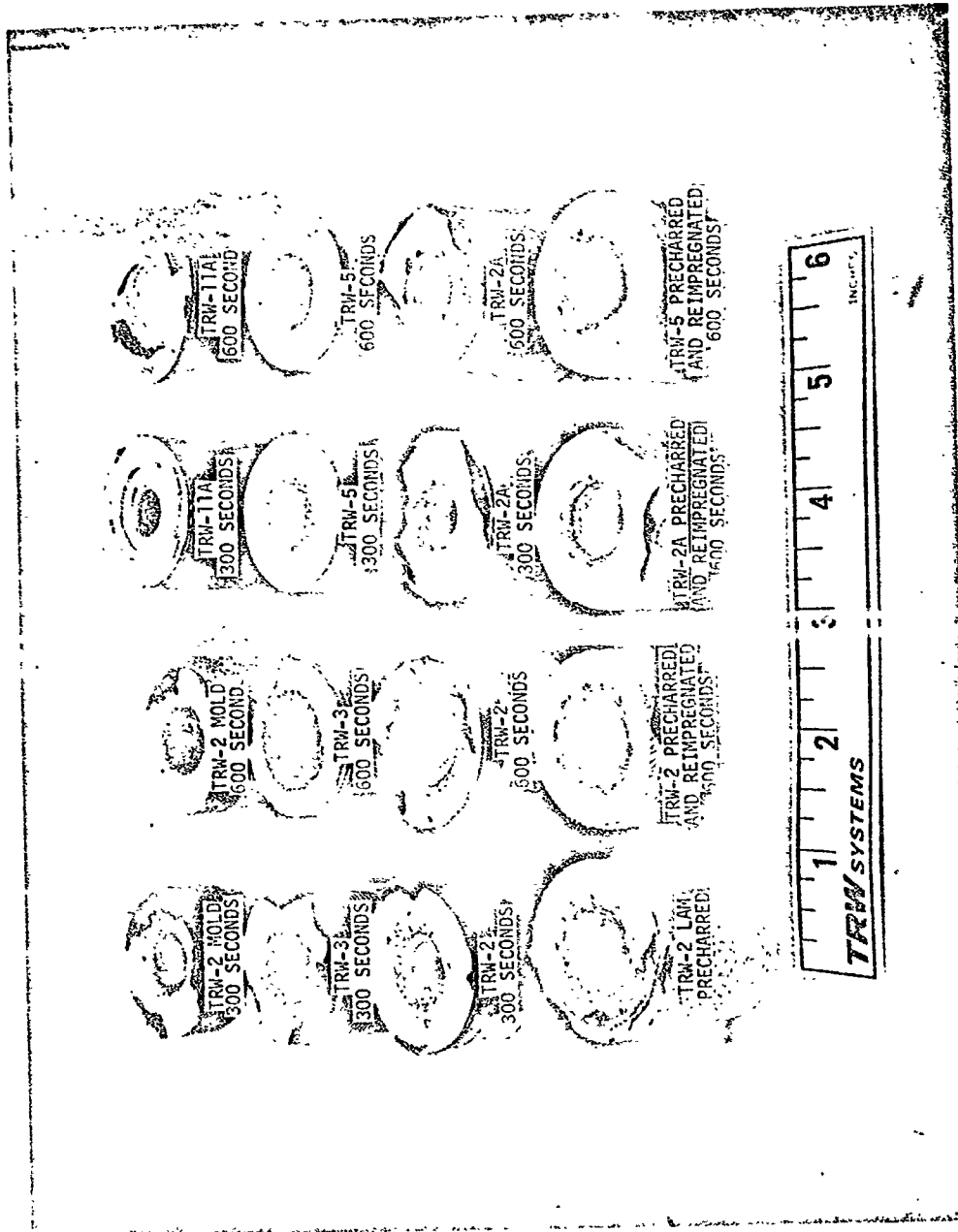


Figure 2-25 TRW Laminate Formulations after Torch Test

Table 2-5. Torch Test Data

Specimen	Specimen Surface Temperature (°C)	Time (Seconds)	Weight Loss		Average Percent Erosion*
			Grams	Percent	
MX-2600	1760	300	11.73	16.4	3.1
MX-2600	1835	600	13.75	19.4	4.8
TRW-2A	1750	300	13.16	20.1	2.0
TRW-2A	1760	600	15.05	22.3	4.9
TRW-2A Precharred and Reimpregnated	1710	600	11.31	17.8	4.7
MXQ-190	1850	300	10.32	16.2	2.2
MXQ-190	1770	600	14.07	19.7	6.4
HRX-211	1900	300	3.19	4.7	6.5
HRX-212	1825	300	2.62	3.9	5.9
Carbitex 100 (SiC mod.)	1750	600	9.18	22.8	34.0
Carbitex 100 (ZrB ₂ mod.)	1960	600	11.27	22.8	43.0
Carbitex 100 (TiB ₂ mod.)	1660	600	11.39	24.6	45.1
Carbitex 100 (W ₂ B mod.)	1650	600	17.16	30.7	74.9
Carbitex 100 (B ₄ Si mod.)	1645	600	9.26	19.4	37.3
Pyrocarb 751	1800	300	2.31	2.8	0.1
Pyrocarb 751	1780	600	5.09	6.7	0.1
TRW-5	1765	300	11.41	15.4	2.2
TRW-5	1770	600	14.29	20.2	9.2
TRW-5 Precharred and Reimpregnated	1770	600	9.67	17.3	7.5
TRW-2	1820	300	12.09	18.1	4.2
TRW-2	1810	600	14.60	21.9	9.1
TRW-2 Mold	1790	300	9.87	14.7	1.9
TRW-2 Mold	1790	600	11.51	20.1	9.0
Quartz Polyimide	1750	300	8.40	12.5	6.7
Quartz Polyimide	1740	600	12.10	17.8	8.0
EC-260 Silica (W mod.)	1850	300	8.24	9.7	5.7
EC-260 Silica (W mod.)	1850	600	17.01	25.0	8.9
Ironsides Resin with Astroquartz	1700	300	10.97	14.1	5.2
Ironsides Resin with Astroquartz	1750	600	13.31	19.6	2.1
C-100-48 Refrasil Low Resin	1940	300	7.82	11.3	12.9
C-100-48 Refrasil Low Resin	1920	600	10.49	15.2	4.4

Table 2-5. Torch Test Data (Continued)

Specimen	Specimen Surface Temperature (°C)	Time (Seconds)	Weight Loss		Average Percent Erosion*
			Grams	Percent	
C-100-48 Refrasil High Resin	1860	300	8.50	12.5	3.8
C-100-48 Refrasil High Resin	1840	600	10.43	15.4	5.5
C-100-28 Refrasil	1790	300	8.48	13.0	4.9
C-100-28 Refrasil	1850	600	12.30	18.7	7.7
TRW-3	1810	300	10.51	16.6	2.3
TRW-3	1815	600	13.06	20.6	11.5
Carbitex 100 (SiC mod.)	1770	300	3.87	9.5	13.8
Carbitex 100 (ZrB ₂ mod.)	1950	300	5.42	10.8	25.0
Carbitex 100 (W ₂ B mod.)	1720	300	9.40	14.0	27.6
Carbitex 100 (B ₄ Si mod.)	1645	300	5.02	10.4	21.8
Carbitex 100 (TiB ₂ mod.)	1660	300	5.41	11.8	31.4
TRW-11A	1650	300	11.67	18.5	5.4
TRW-11A	1660	600	16.00	25.1	16.3
TRW-2 Precharred	1880	600	7.10	12.1	5.2
TRW-2 Precharred and Reimpregnated	1780	600	10.31	15.8	5.7
Type C W-3Re Reinforced Zirconia	1980	600	10.20	--	--
Type A W-3Re Reinforced Zirconia Bonded Zirconia	2000	600	--	--	--
Zirconia Bonded Zirconia	2030	300	--	--	--
Zirconia Bonded Zirconia	2020	600	--	--	--
Carbone Graphite (SiC coated)	1610	600	0.46	0.153	--
ZrB ₂ V	1760	600	--	--	--
ZrB ₂ VIII	1760	600	--	--	--
Porous Tungsten with L13 Treatment	1620	300	--	--	--
Porous Tungsten with L13 Treatment	1720	600	--	--	--
JTA Graphite	1800	600	2.59	0.53	--
Crystar (SiC mod.)	1640	600	--	--	--
Graphite Phenolic (ZrB ₂ mod.)	2020	600	13.18	16.0	4.2
SiC Powder 1200 Mesh Graphite Fiber Reinforced C-1000 Phenolic Precharred	1815	600	5.66	8.4	13.8

Table 2-5. Torch Test Data (Continued)

Specimen	Specimen Surface Temperature (°C)	Time (Seconds)	Weight Loss		Average Percent Erosion*
			Grams	Percent	
TRW-2A Precharred and Reimpregnated**	1600	600	10.15	15.6	5.0
TRW-5 Precharred and Reimpregnated**	1700	600	7.41	10.7	1.7
MX-2600**	1750	300	10.33	14.6	0.82
MX-2600**	1740	600	13.43	19.0	1.17
EC-260 Resin with Astroquartz	1750	300	9.35	13.4	2.7
EC-260 Resin with Astroquartz	1745	600	12.96	19.0	3.7

* Average percent erosion is obtained by taking the average of the absolute percent diametrical changes throughout the length of the hole in the test specimen. Thus, average erosion = $\frac{\sum \Delta x_i}{N}$ where x_i is percent erosion at location i , and N is the number of measurements.

** These specimens were tested using a gas mixture of 50 CH₄ to 50 O₂ by volume.

--Weight loss or erosion less than 0.1%.

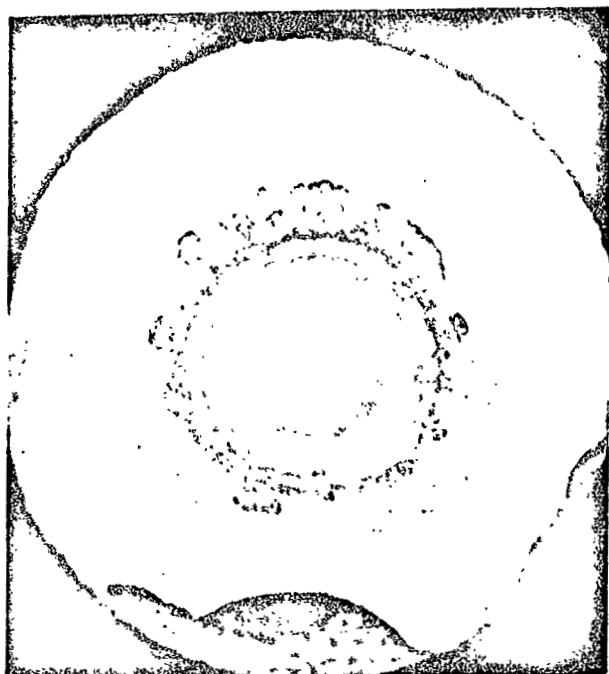


Figure 2-26 TRW-2A Precharred and Reimpregnated 600 Second Test (2X)



Figure 2-27 TRW-5 Precharred and Reimpregnated 600 Second Test (2X)

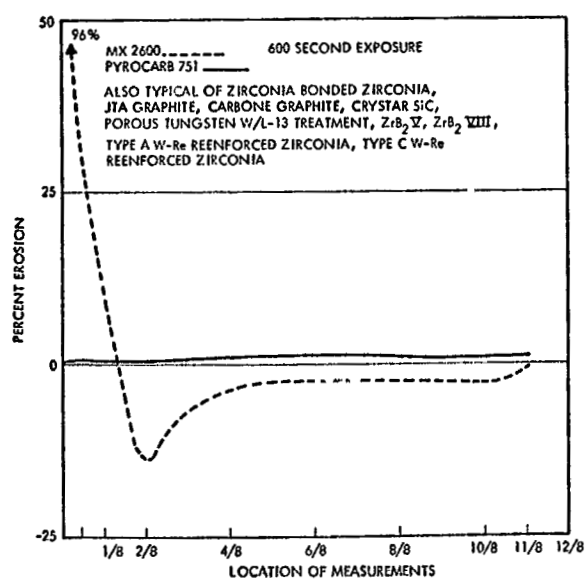


Figure 2-28. Erosion Profile of "Hard Throat" Materials

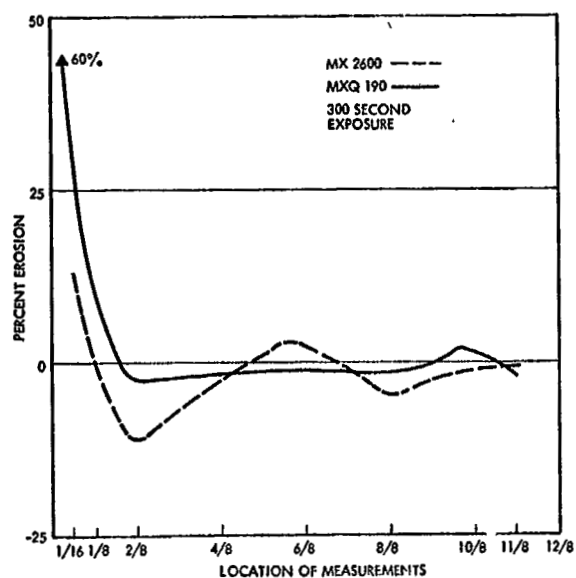


Figure 2-29. Erosion Profile of MXQ-190 300 Second Test

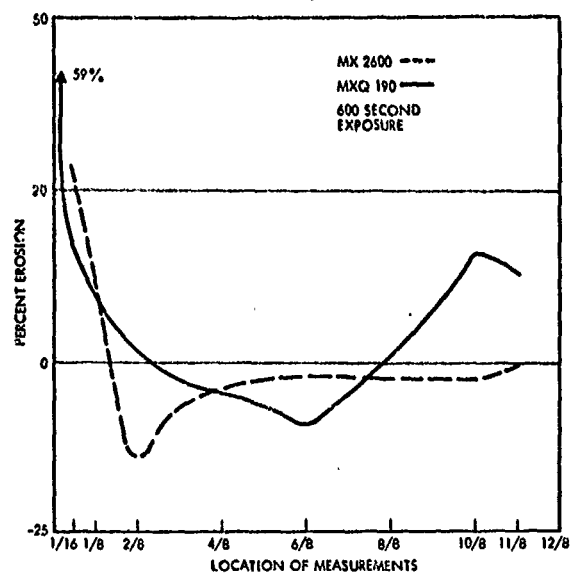


Figure 2-30. Erosion Profile of MXQ-190
600 Second Test

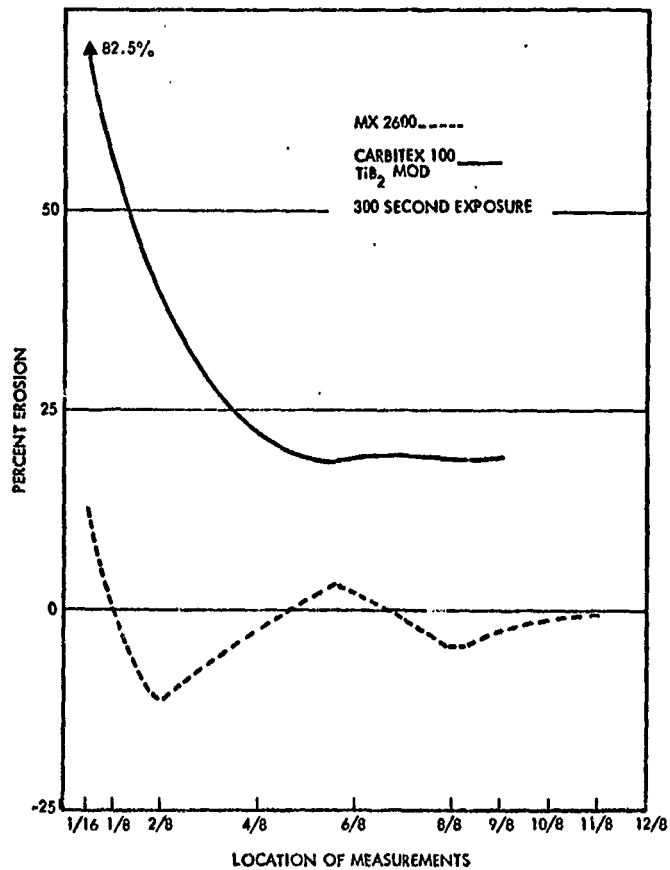


Figure 2-31. Erosion Profile of Carbitex 100
(TiB₂) 300 Second Test

to form silicon carbide. Therefore, two of the most promising materials, TRW-2A and TRW-5 (see Table 2-5), were retested, along with the baseline MX-2600 material, with a fuel-rich flame. TRW-2A was a phenolic laminate containing two plies of quartz fabric (instead of the silica fabric of TRW-2) per ply of graphite fabric.

The effect of gas atmosphere on weight loss and erosion from torch testing appears to be significant for certain composite formulations. A lower weight loss was obtained in TRW-2A and TRW-5, but not in MX-2600, when a fuel-rich mixture was used. However, a lower erosion rate was also evident by comparing the dimensional changes in all specimens except TRW-2A tested at fuel-rich gas mixtures.

X-ray diffraction analyses were performed on several specimens after torch testing to determine if silicon carbide had formed. The following are the results of these analyses:

<u>Specimen</u>	<u>Test Conditions</u>	<u>Silicon Carbide Detection</u>
MX-2600	LMDE Throat (duty cycle)	Yes
MX-2600	Torch Test (40 CH ₄ - 50 O ₂ - 10 NO)	No
MX-2600	Torch Test (50 CH ₄ - 50 O ₂)	Yes
TRW-2A	Torch Test (50 CH ₄ - 50 O ₂)	Yes

It is interesting to note that a less oxidizing atmosphere was favorable for the formation of silicon carbide which formed preferentially on the surface in contact with the flame.

2.3.4 Torch Test Conclusions and Recommendations

On the basis of erosion or dimensional change measurements of torch tested specimens, it was concluded that only hard throat materials such as JTA, SIC, W-reinforced zirconia, zirconia, and coated carbon/graphite materials such as SiC-coated Carbone graphite and Pyrocarb 751 can withstand the simulated LMDE throat environments with little or no erosion. However, poor thermal shock resistance in some of these materials and insufficient development in others may preclude the selection of these materials for full-scale evaluation.

Of the ablative type of resin-base materials, TRW-2A, quartz/Ironside resin, MXQ-190, and MX-2600 showed about equal erosion resistance in torch tests. On the basis of 300-second testing, several formulations such as TRW-2A and MXQ-190 appear to have an edge over MX-2600. Full-scale engine tests were recommended to verify these conclusions.

Some indications were shown in test results that precharring or precharring followed by resin reimpregnation is effective in improving the erosion performance of the precursor resin-base material. Encouraging results shown by precharred TRW-2 and TRW-5 formulations warranted verification by engine testing.

3. LMDE CHAMBER THERMAL ENVIRONMENT

3.1 PROGRAM OBJECTIVE

The objective of Phase I of the subject study effort was to define and characterize the internal combustion chamber environment and to determine methods to minimize the severity of the environment to which the LMDE ablative chamber is subjected. Heat transfer studies and tests have been conducted with qualification design head end assemblies to evaluate the thermal environment. Heat transfer rigs and measuring techniques were developed and tested in order to evaluate heat transfer with the Oval B chamber contour, variations in the internal chamber contour (modified contours), injector flow distribution, barrier cooling flow variation, and engine thrust level.

3.2 DEFINITION OF EXPERIMENTAL DATA REQUIREMENTS

Specially designed and instrumented thrust chamber hardware was utilized for measurement of heat transfer to local areas of the chamber wall. These measurements were taken during firings at various thrust levels to characterize the chamber thermal profile, both axially and circumferentially. The basic heat-sink chamber hardware was employed for short duration firings at selected engine operating conditions. The local wall heat transfer data was obtained utilizing transient surface temperature techniques and data reduction procedures described in detail in Section 3.5.3. The water-cooled chamber design provided local wall heat flux data during long duration/variable thrust level engine operation. The heat flux was readily determined from the measured water flow rates and temperature rise under chamber equilibrium operating conditions.

Streak chambers were tested at selected fixed thrust settings to obtain throat erosion patterns. These throat erosion data were compared and correlated with the thermal environment data from the heat-sink and water-cooled chamber tests to arrive at the variation of erosion with heat flux. In addition, ablative (MX 2600) chambers were duty cycle tested to provide throat erosion patterns. These data were also compared with the heat flux data to provide the erosion/thermal environment correlations.

The diagram of Figure 3-1 depicts the data flow achieved in the test programs to arrive at the thermal environment/erosion correlations presented herein.

3.3 SIGNIFICANT CONCLUSIONS AND INFORMATION

- The heat sink tests showed that the hot wall thermal environment does not exhibit constant recovery temperature/heat transfer coefficient levels for wall temperatures from ambient to 1800°F.
- The use of theoretical film coefficients in conjunction with the heat flux data provides the most obvious and currently acceptable means to derive recovery temperature from the CIC test rigs.
- The throat recovery temperatures vary from 4000°F to 4650°F over the LMDE Qual B thrust range. (See Section 3.5.4.4.)
- Convective heat transfer coefficients in the Qual B chamber barrel section were shown to be up to 5.5 times theoretical values.
- At low thrust levels the barrel section heat flux was greater than the throat section.
- The throat heat transfer coefficients appeared to vary from 1.0 to 1.1 times the theoretical values. (See Section 3.6.1.2.)
- A correlation of the erosion and thermal environment was accomplished with a streak and an ablative chamber at 25% thrust. The data showed the erosion to vary directly with heat flux changes.
- At thrust levels of 50%, 65% and FTP, heat flux changes generally produced proportionate changes in erosion depth.
- A throat heat flux reduction of 40% was accomplished in going from a Qual B contour to a Contour "A" with a heat sink throat. Duty cycle tests of the same two configurations with ablative throats showed Contour "A" to be more eroded than Qual B. Since the heat sink tests were conducted only at FTP the effects of the duty cycle erosion sequence could not have been predicted. Also, streak chamber tests at FTP for both chamber contours showed a reduction in erosion for Contour "A", thus substantiating the heat flux reduction at the fixed thrust point.

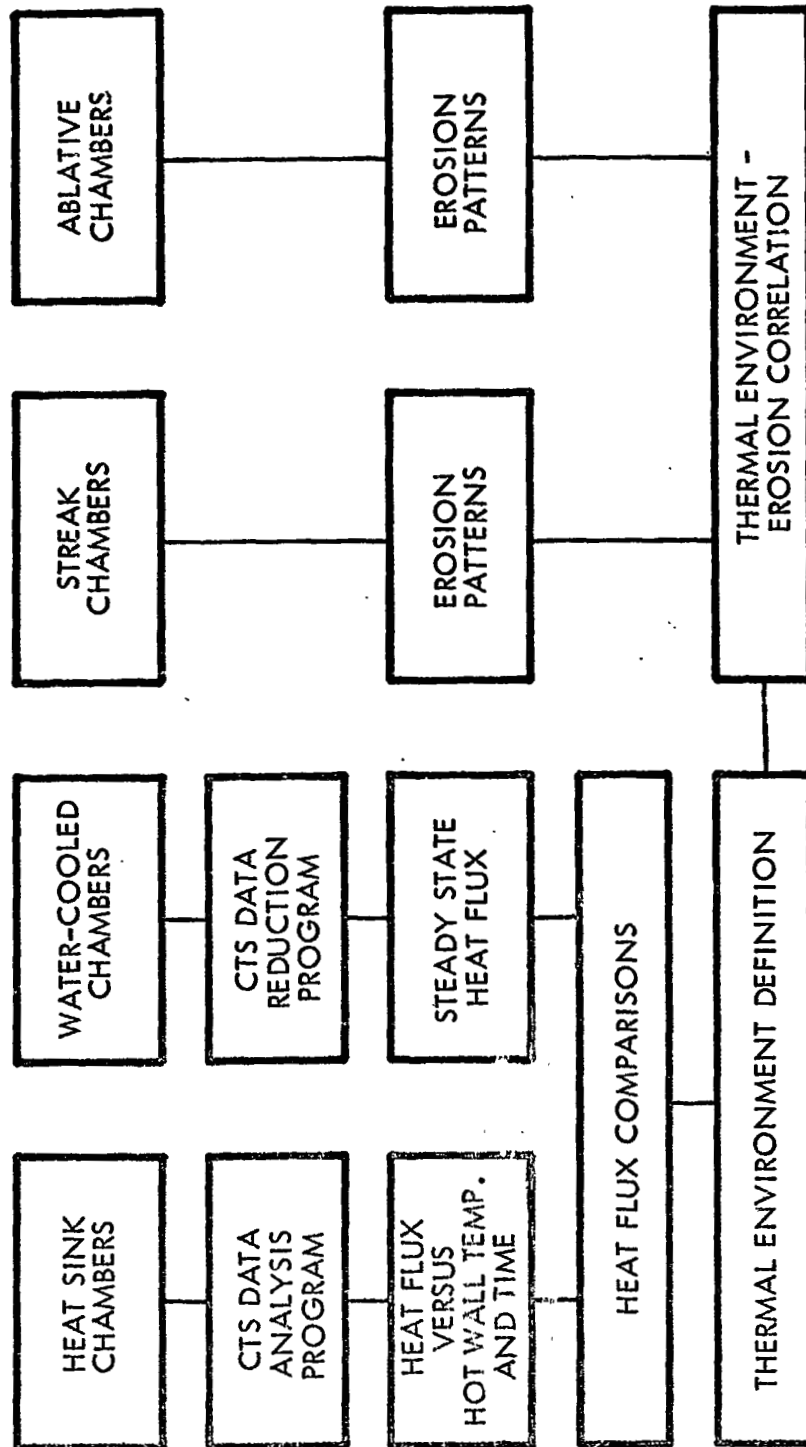


FIGURE 3-1 DATA ANALYSIS FLOW CHART

- The effects of heat flux on LMDE throat erosion was deduced to be only part of a more complex problem. The erosion is caused by several mechanisms of temperature, shear, oxidation, melt viscosity, and other properties. But perhaps the most significant mechanism is one of thermochemical reactions occurring at the wall. The interaction of a complex two-zone combustion flow field in which primary and secondary oxidizer streams react with the fuel sheet and the barrier cooling flow points up the fact that the zone nearest the wall at the throat is significantly detrimental to the integrity of MX2600 ablative material during the present LMDE duty cycle firing sequence.

3.4 ANALYTICAL GAS-SIDE HEAT TRANSFER COEFFICIENTS

The gas-side heat transfer to the chamber wall is a function of the propellant properties, chamber pressure, and engine geometry. The most important heat transfer mode for the LMDE chamber is that of forced convection.

There are several forced convection correlations currently in use for rocket thrust chambers. These include both boundary layer and pipe flow equations. The most commonly used, easily applied closed-form pipe flow solution is that of Bartz's simplified analysis technique:

$$h_g = \left[\frac{c}{D^{*0.2}} \left(\frac{\mu^{0.2} C_p}{P_r^{0.6}} \right) \left(\frac{P_c g}{C^*} \right)^{0.8} \right] \left(\frac{A^*}{A} \right)^{0.9} \sigma$$

Using the Bartz reference temperature

$$T_{REF} = \frac{T_g + T_w}{2}$$

σ is given by

$$\sigma = \frac{1.0}{\left[\frac{1}{2} \frac{T_w}{T_0} \left(1 + \frac{\gamma-1}{2} M^2 \right) + \frac{1}{2} \right]^{0.8-\omega/5} \left[\left(1 + \frac{\gamma-1}{2} M^2 \right)^{\omega/5} \right]}$$

where: ω = slope of the gas viscosity - temperature data
 γ = ratio of specific heats

An investigation of the LMDE combustion gas property variations from available rocket chemistry data was made. The current LM Descent Engine injector exposes the chamber to a two-zone (on a simplified basis) combustion flow field; the zone nearest the wall has a mixture ratio which varies with length of travel and, quite possibly, with wall temperature. The range of wall mixture ratios investigated should adequately represent the maximum probable for the LMDE chamber.

The property variations taken from rocket chemistry data for $N_2O_4/50-50 N_2H_4$ - UDMH were as follows:

O/F Ratio	Cp (Btu/lb-°F)	Pr	$(lb_m/in-sec)$	γ
0.8	0.5785	0.7869	2.65×10^{-6}	1.277
1.2	0.5431	0.8013	3.29×10^{-6}	1.277
1.6	0.5109	0.8081	3.65×10^{-6}	1.247
2.0	0.4841	0.8108	3.82×10^{-6}	1.235
2.2	0.4728	0.8114	3.87×10^{-6}	1.228
	* -0.2%	-0.1%	-0.8%	0.1% for $P_c=30$ psia, O/F = 1.6
	* 0.5%	0.3%	3.9%	-0.4% for $P_c=300$ psia, O/F = 1.6

* Percent variations shown correspond to property differences at the specified conditions (P_c and O/F) compared to $P_c = 50$ psia, O/F = 1.6. The variations shown are typical of that experienced over the entire mixture ratio range.

These results show that the theoretical gas property variations for mixture ratios from 0.8 to 2.2 are insignificant and would have little effect on the Bartz convective heat transfer coefficients in the LMDE chamber and throat.

A full range of LMDE chamber-throat heat transfer coefficients as a function of O/F P_c , wall temperature, and chamber location was generated by use of the Bartz simplified analysis technique in order to facilitate the analytical and experimental heat transfer studies planned in support of the program. The results of this analysis are shown in Figure 3-2.

3.5 HEAT SINK CHAMBERS

The following subsections describe the heat sink chambers utilized in the heat transfer tests for the chamber/injector compatibility improvement study. Specifically, the discussion covers the design, analysis, and subsequent tests and results for the heat sink chamber and throats utilized throughout the program.

3.5.1 Chamber Design

The Qual B heat sink thrust chamber design as shown in Figure 3-3 employed mild steel cylindrical chamber sections and a contoured throat section. A 1.5 inch thick chamber wall was established such that the backwall (exterior) would not experience a temperature increase during the planned short duration firing tests. This restriction allows the use of a single hot side wall temperature measurement as opposed to a temperature gradient measurement that would require chamber outside surface temperature measurements opposite each hot wall surface thermocouple. Generally, the heat transfer prediction techniques that yield local heat flux require both hot and cold wall temperature responses as a function of time over the engine firing duration. Elimination of the cold wall temperature responses in the transient analysis (1) reduces the required data acquisition volume, and (2) reduces data reduction and analysis time. Surface temperature probes (per Reference 5) were utilized to obtain both axial and circumferential temperature data in the chamber and throat areas as shown in the photograph on Page 3-10. The thermocouples were spaced as shown in Figure 3-4 to measure regions on and between barrier cooling holes. The angular orientation of the chamber relative to the flow control valves was, of course, variable so as to be able to line up the groups of T/C's at station 3 and 7 with regions of high erosion tendencies as shown by streak and ablative chamber tests. Reference 3 further details the hardware requirements and design criteria for the heat sink chamber.

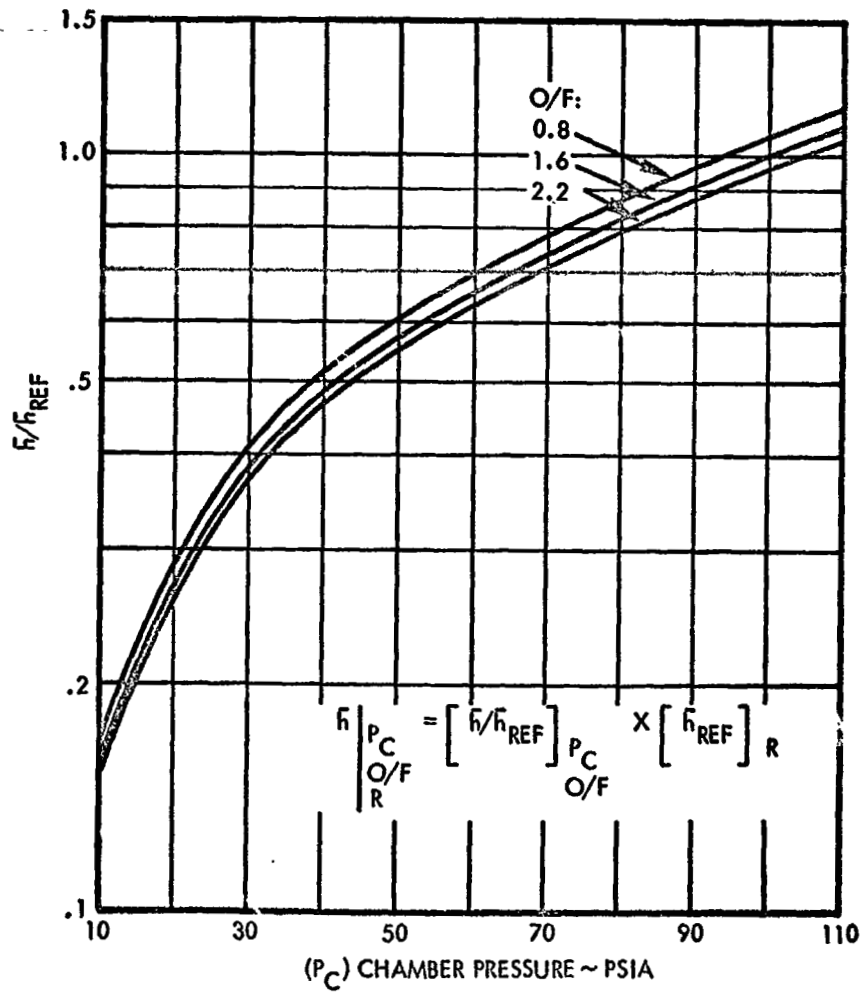
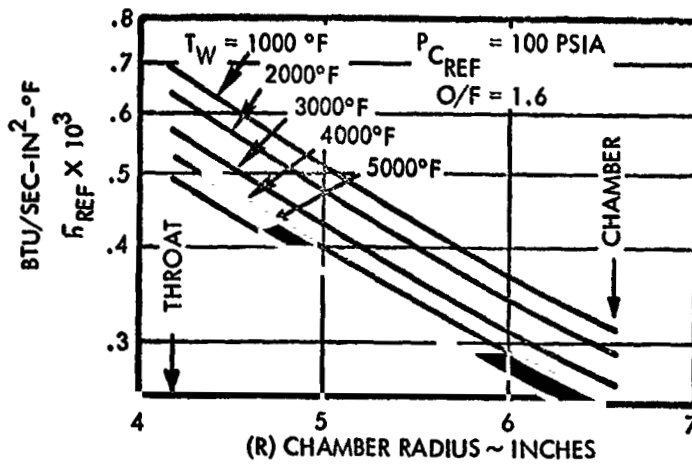


FIGURE 3-2 LMDE CHAMBER-THROAT HEAT TRANSFER COEFFICIENTS

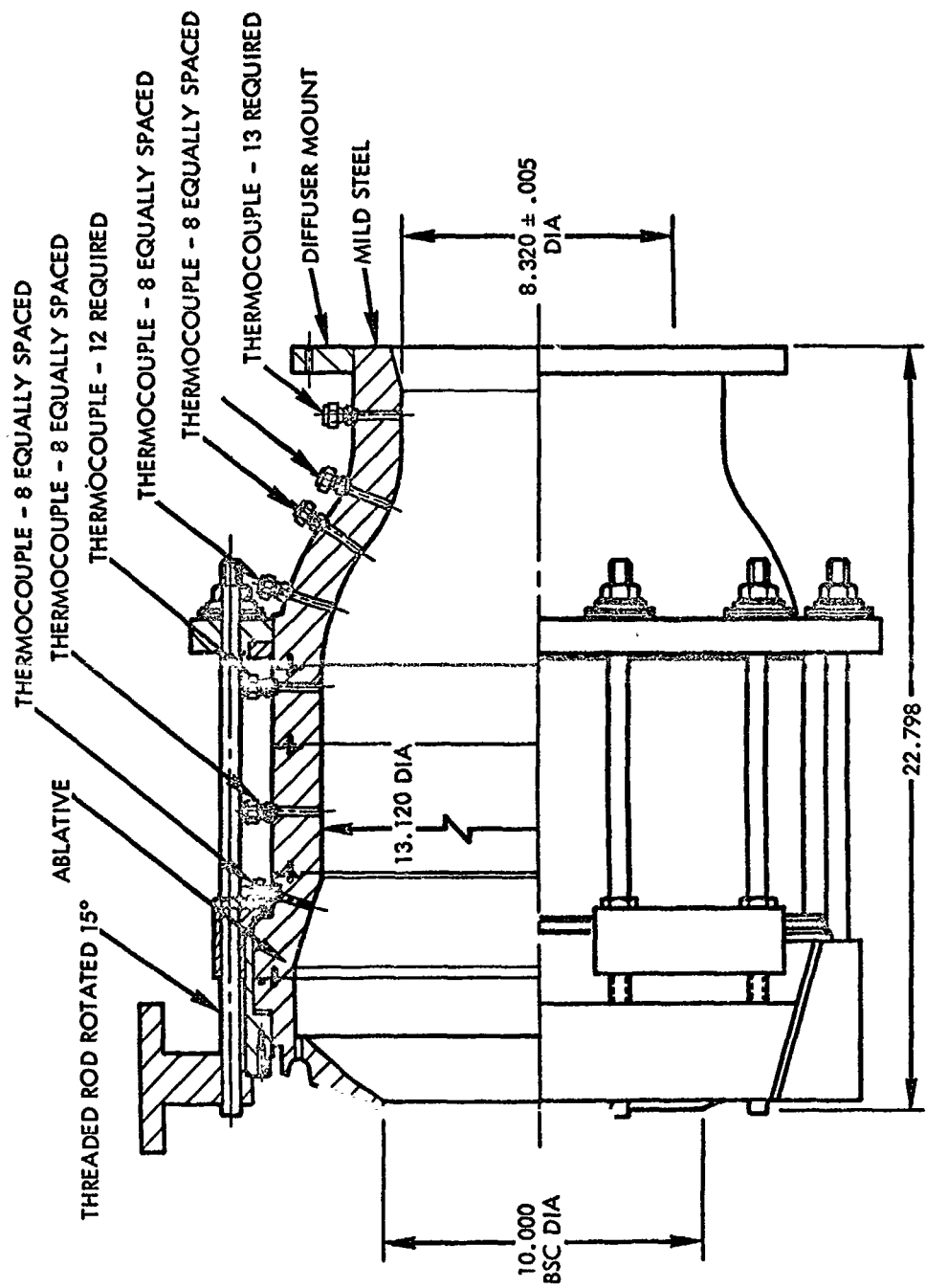


FIGURE 3-3 WORKHORSE HEAT SINK THRUST CHAMBER DESIGN

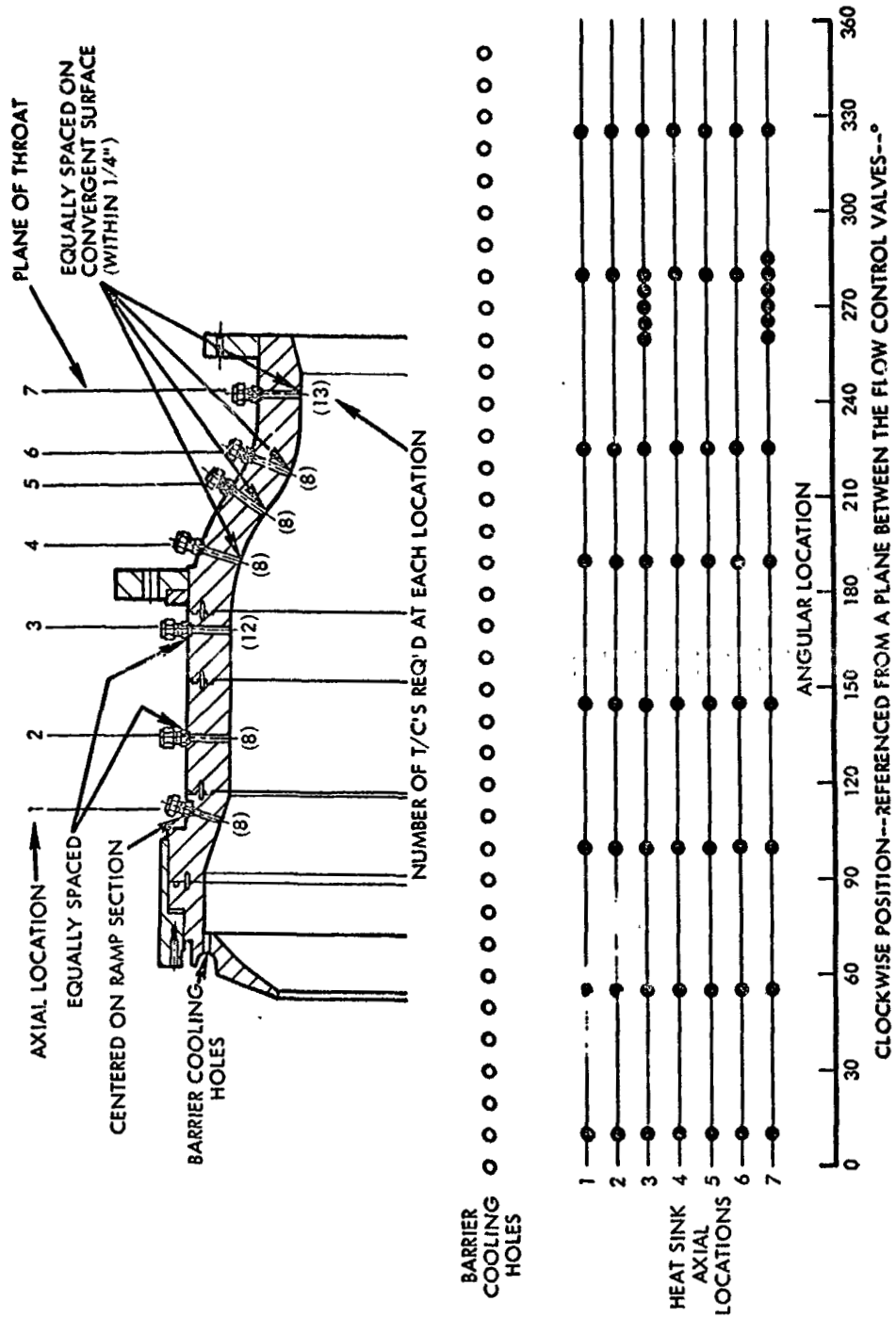
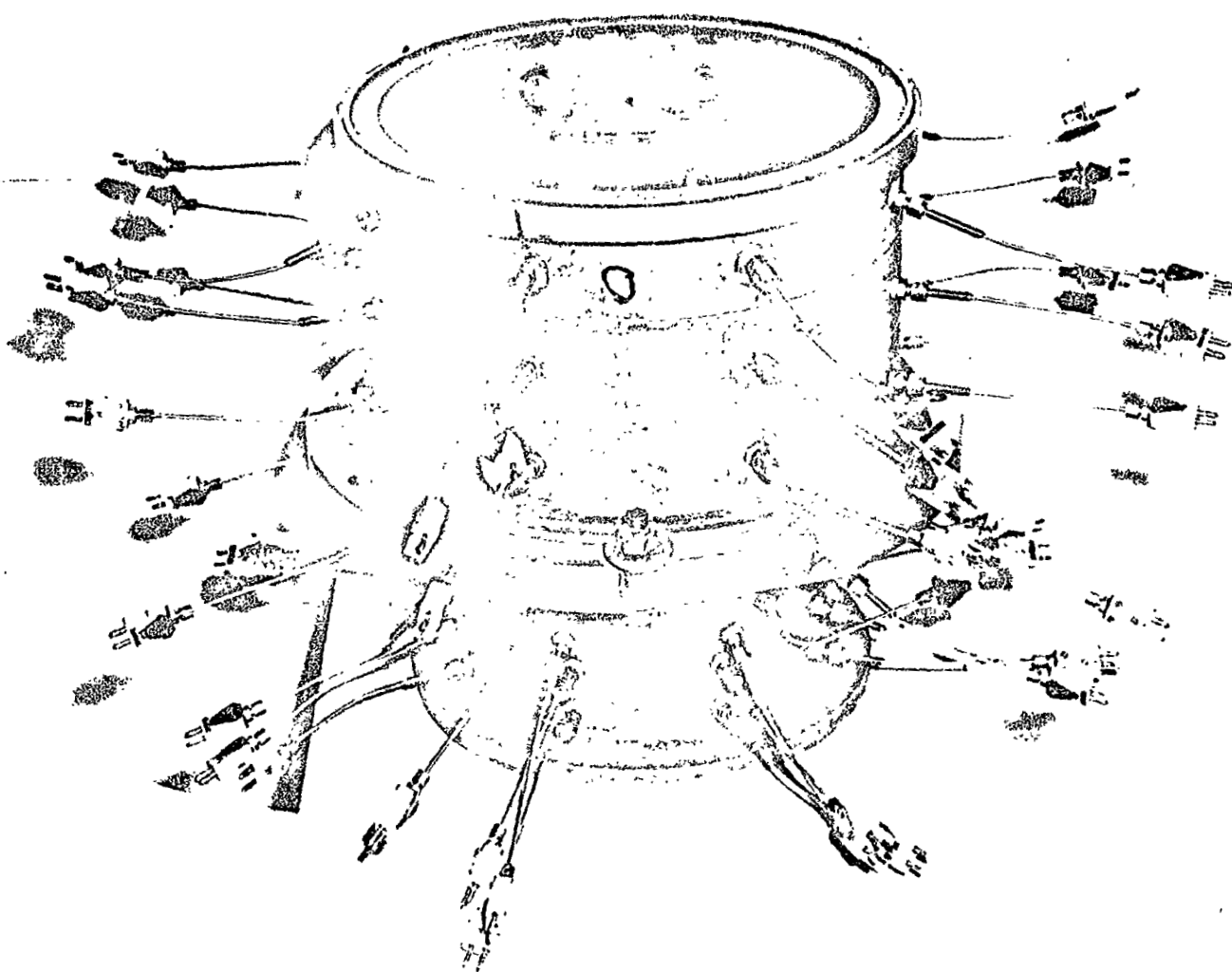


FIGURE 3-4 HEAT SINK CHAMBER THERMOCOUPLE AXIAL AND ANGULAR LOCATION

11602-6039-R0-67



HEAT SINK CHAMBER PHOTOGRAPH

3.5.2 Analytical Thermal Analysis of Heat Sink Throat

An analysis was performed on the heat sink thrust chamber to determine the probable thermal operating characteristics. The analysis was conducted at the throat station utilizing the heat sink thermal model as shown in Figure 3-5. The internal and external surface temperatures were determined as functions of time, chamber pressure, and effective gas recovery temperature. Previously, limited heat transfer data indicated average effective gas temperatures varied from approximately 3500 to 5000°F over the range of operation (thrust level) of the LMDE chamber using Qual B head end assemblies. Therefore, this parametric analysis encompasses all possible heat loads anticipated during actual engine operation.

The analytical results shown in Figures 3-6, 3-7, and 3-8 were used as guidelines in establishing test duration limitations. The two prescribed internal (hot wall) temperature limits of 1400 and 1700°F were adequate to ensure chamber integrity and overall durability while providing the desired local wall environmental data. Temperatures up to 2000°F could be tolerated provided the stay time at the higher temperature was of short duration so as not to cause local oxidation streaking across the thermocouples. See Reference 3 for additional analytical temperature data.

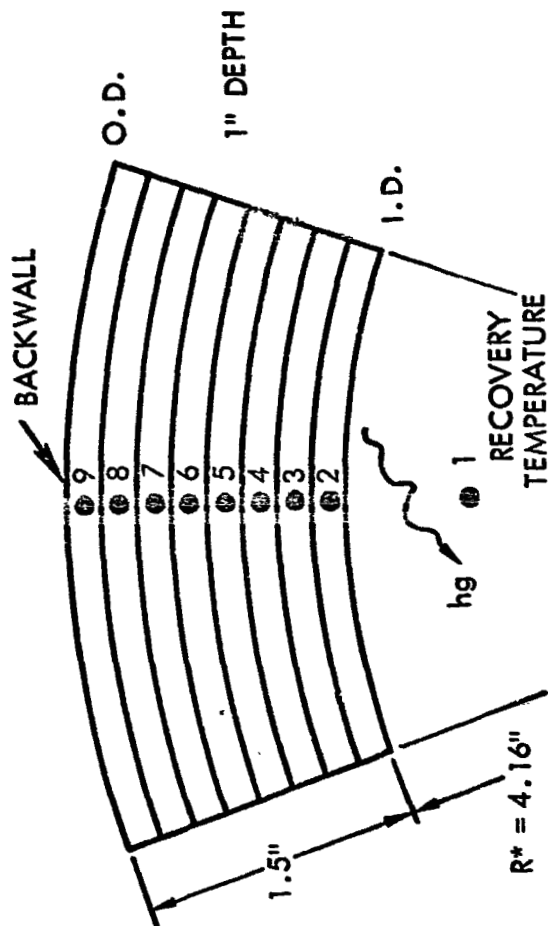
3.5.3 Data Reduction and Analysis Technique

The method of data reduction for the heat sink thermocouple test data was selected based on the following desired objectives:

- Fast turnaround time from test to reduced data in engineering form.
- Processing of the data be accomplished at the CTS onsite computer.
- Consistent analysis technique.
- Inexpensive data reduction.

An approximate solution of the temperature diffusion equation was selected for use in the on-line data reduction program.

- THROAT LOCATION
- ADIABATIC BACKWALL
- WALL THERMAL PROPERTIES OF
SAE 1020 LOW CARBON STEEL
- ONE-DIMENSIONAL RADIAL CONDUCTION



hg VALUES FROM BARTZ SIMPLIFIED ANALYSIS TECHNIQUE

FIGURE 3-5 THERMAL MODEL

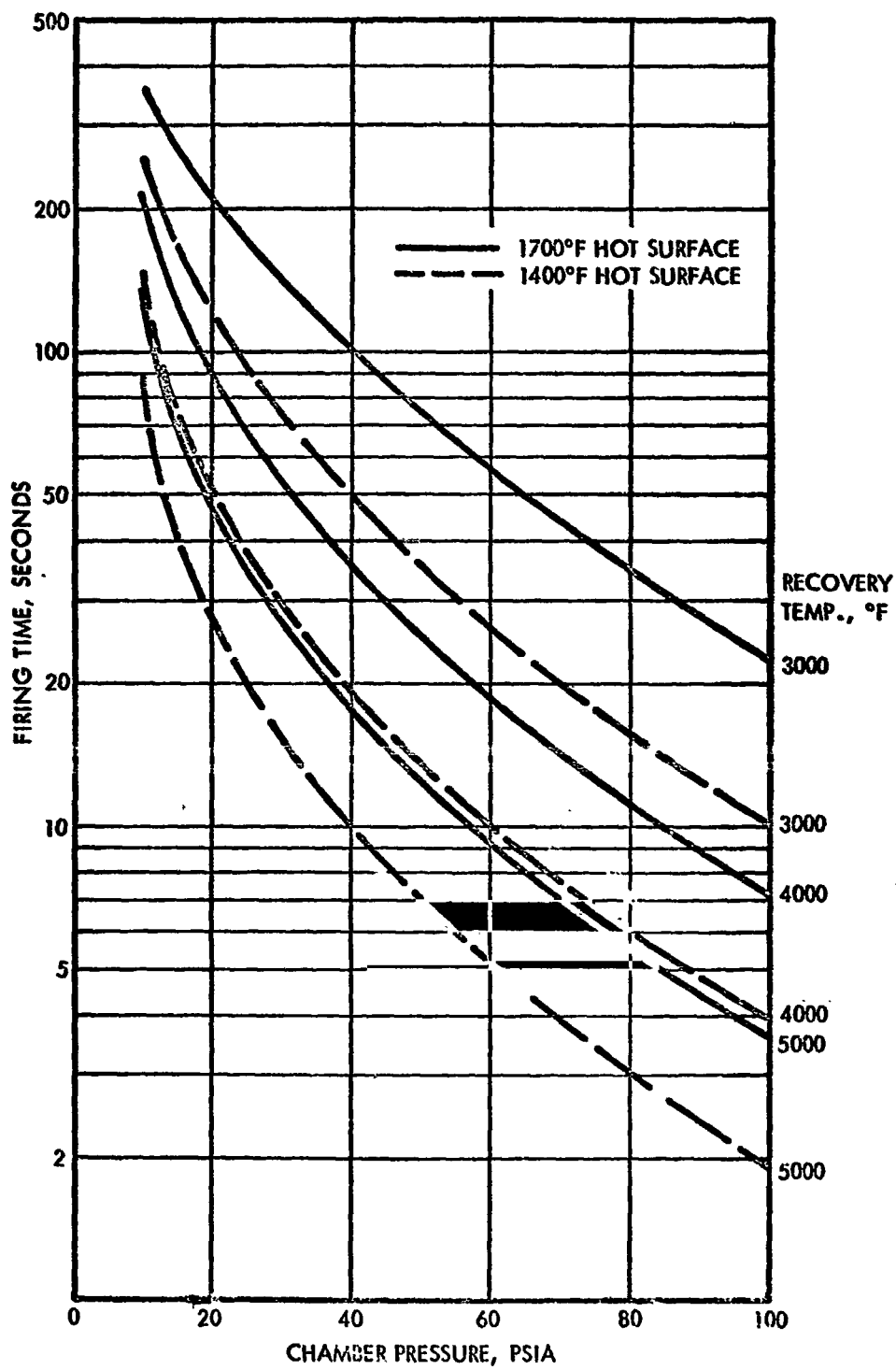


FIGURE 3-6 HEAT SINK CHAMBER PERMISSIBLE FIRING DURATION

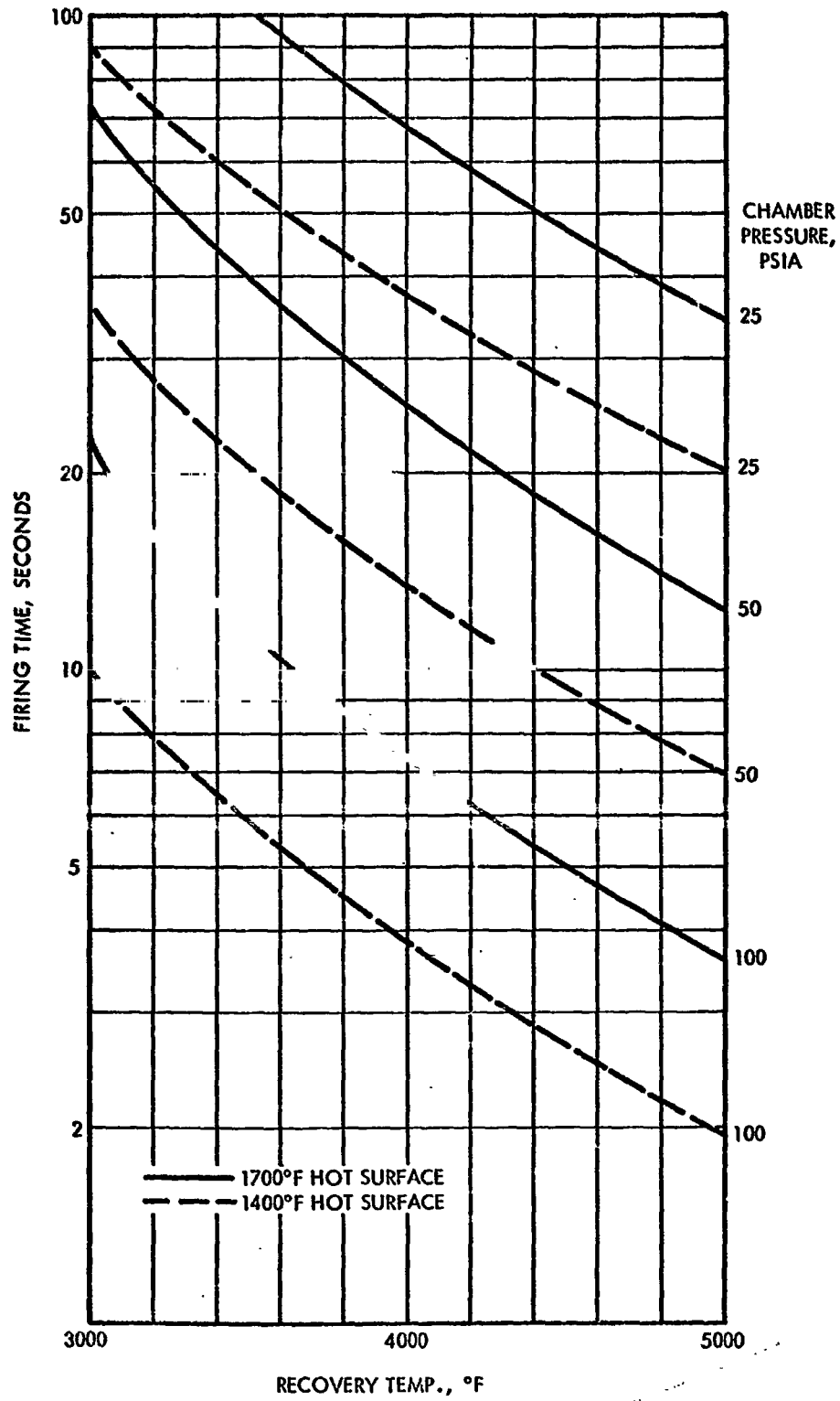


FIGURE 3-7 HEAT SINK CHAMBER PERMISSIBLE FIRING DURATION

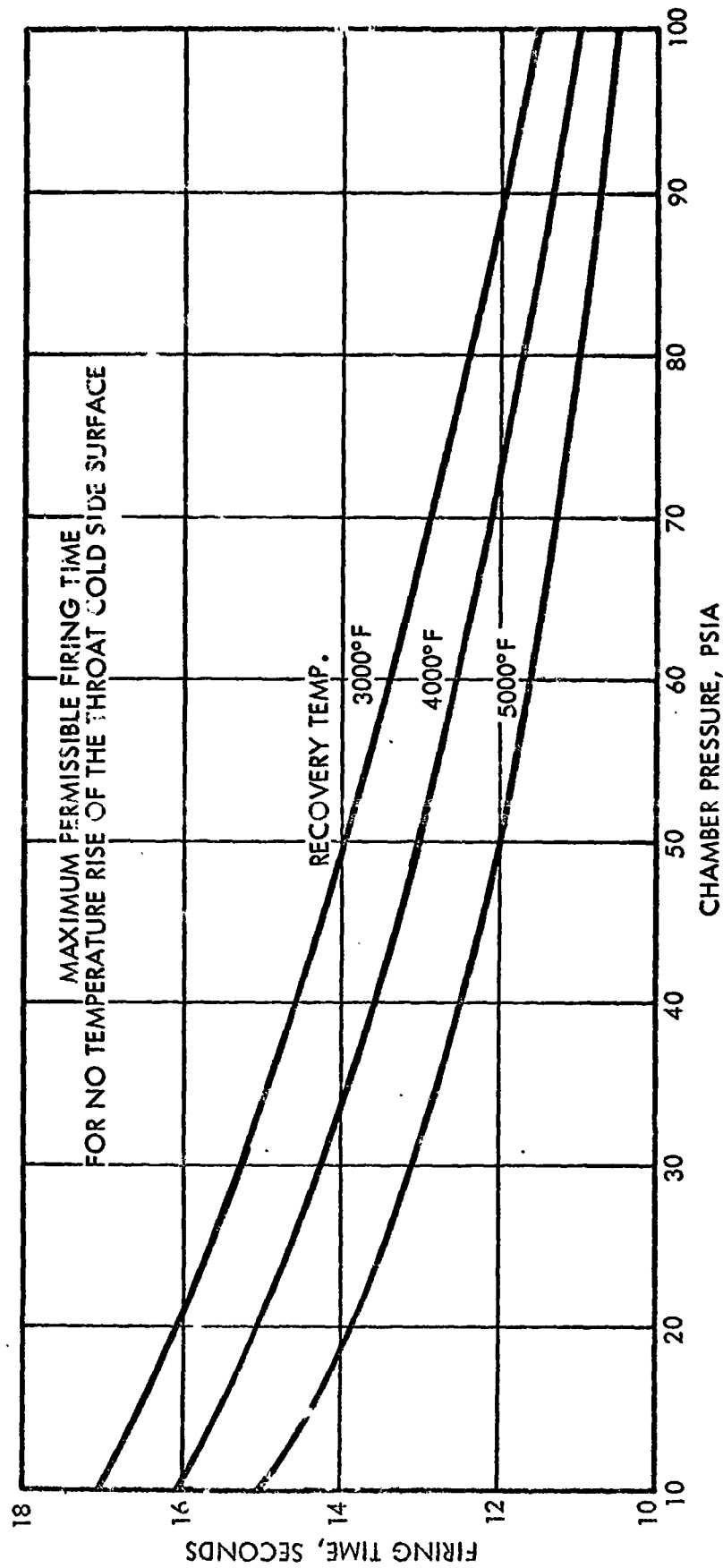


FIGURE 3-8 HEAT SINK CHAMBER MAXIMUM PERMISSIBLE FIRING TIME
FOR NO TEMPERATURE RISE OF THE THROAT COLD SIDE SURFACE

The solution is of the form,

$$\frac{T_w - T_o}{T_R - T_o} = \frac{\frac{h_g}{k} \sqrt{\frac{3}{2} \alpha \theta}}{1 + \frac{h_g}{k} \sqrt{\frac{3}{2} \alpha \theta}} \quad (1)$$

where:

- T_w = wall temperature at time θ
- T_o = initial wall temperature
- T_R = recovery temperature
- h_g = convective film coefficient
- k = material thermal conductivity
- α = material thermal diffusivity
- θ = time

This solution requires the following assumptions and boundary conditions between any two consecutive time points:

- Constant recovery temperature
- Constant film coefficient
- Constant wall thermal properties
- Semi-infinite wall body

This simplified solution was found to yield comparable heat flux results to the more elaborate numerical solutions of the temperature diffusion equation which requires complete wall interior mesh point temperature distribution calculations with iterative techniques, convergence restrictions and stability criteria in order to arrive at the hot side surface thermal gradient and the heat flux. Studies showed that the approximate technique yields the same heat flux values (within $\pm 5\%$) as the more elaborate numerical solution.

Rearranging Equation (1),

$$T_w = \frac{\sqrt{\theta}}{A + B \sqrt{\theta}} + T_o \quad (2)$$

where,

$$A = \frac{k \sqrt{2/3\alpha}}{h_g (T_R - T_o)} \quad (3)$$

and,

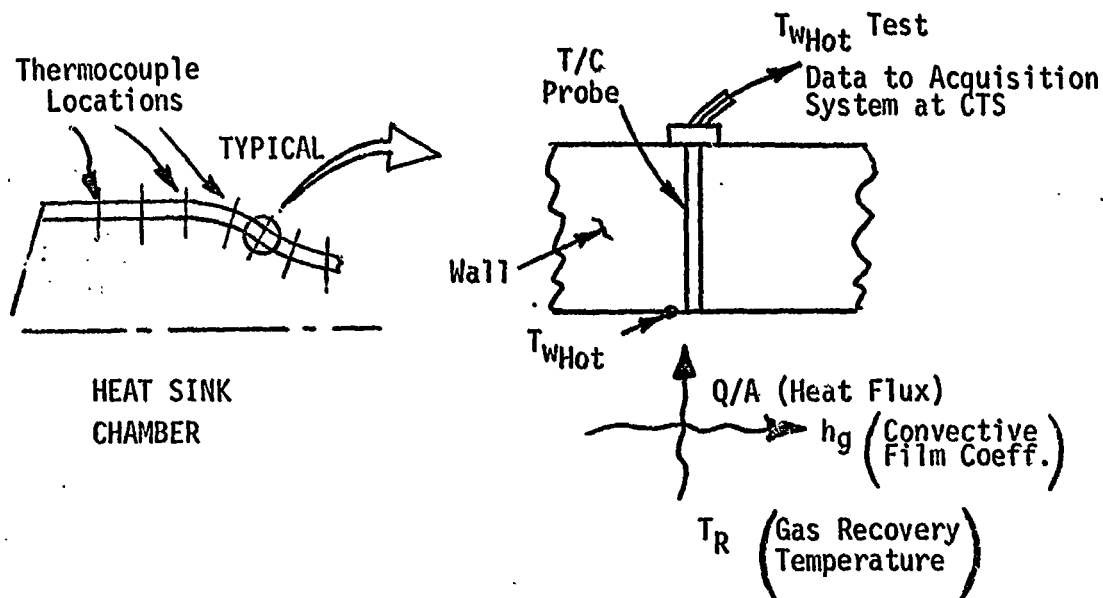
$$B = \frac{1}{T_R - T_o} \quad (4)$$

Application of Equation (2) to analytical wall temperatures showed that this curve form yielded the best fit to the analytical solutions when compared with four other basic curve fit forms. In addition the curve form represented by (2) was compared with test data from steel heat sink chambers and found to yield the most accurate fit over the other basic forms.

From the basic form of Equation (2), it is apparent that the components of heat transfer, namely h_g and T_R , are readily determined once the constants A and B of Equations (3) and (4), respectively, are calculated. This is simply accomplished by imposing a least squares curve fit to the raw wall temperature-time data of the same basic form as Equation (2) and computing the constants A and B.

The Heat Sink Rig Data Analysis Program (HEAT) was written (Reference 13) and implemented on the SDS 930 computer located at CTS to provide the required on-line data reduction capability.

Temperature data from each thermocouple for each test is recorded at the Capistrano Test Site Control Center.



The recorded temperature data were reduced at CTS on the General Data Reduction Processor. This provided a listing of engine performance data and raw wall temperature data from the thermocouples in engineering units. It also writes a FORTRAN readable data tape which contains the raw wall temperature (T_w) and time (θ) measured from the fire switch signal. At this point the data is reviewed and the control data cards set up for HEAT. HEAT is then loaded and executed.

In execution HEAT performs the following functions:

- The data tape is read; temperature data for time θ are found and stored.
- A least squares curve fit is computed for each thermocouple using all data on the tape for time between θ_1 and θ_N .
- The data tape is rewound and the statistics of each curve fit (index of determination, standard error of estimate, and maximum residual) are computed.

- Using the assumptions of a semi-infinite wall, constant gas recovery temperature (T_R) and constant convective heat transfer coefficient (h_g) for each thermocouple and the temperature curve fit, the heat transfer equations are numerically solved for heat flux (Q/A), recovery temperature, and convective coefficient at the N time points between θ_1 and θ_N .
- The computed information is printed out in tabular form.

Complete documentation of the HEAT program is contained in Reference 13.

3.5.4 Qual B Baseline Thermal Environment Characterization

The Qual B heat sink chamber configuration was subjected to fifteen (15) firings between 19 December 1968 and 28 January 1969. Testing was conducted on HEA 023 in the TRW VETS facility. The total reactive time accumulated was in excess of 350 seconds consisting of 8 tests on the initial build and 7 tests on the second build. The full LMDE thrust range was covered in the heat sink tests, namely FTP, 65%, 50%, 25%, and 10%.

3.5.4.1 Heat Sink Chamber Test Summary

The Qual B baseline heat sink chamber tests completed during the characterization test program are presented in Table 3-1.

3.5.4.2 Temperature Data

Typical temperature data from the heat sink chamber are shown in Figures 3-9 and 3-10. These plots show the axial and circumferential variation of hot side wall temperature as a function of thrust level on a common basis of time after fireswitch. Other typical transient temperature traces are shown in Reference 26. These temperature data were used to calculate the heat flux levels throughout the chamber utilizing the HEAT data reduction program.

TABLE 3-1. HEAT SINK CHAMBER TESTS

Date	Test No.	Thrust	Duration* (sec)	Mixture* Ratio	$\frac{\Delta P_{IO}}{\Delta P_{IF}}$	Comments
12-19-68	VA1-673	FTP	5.0	1.58	5.00	Initial build with 65 uncoated T/C's; chamber oriented as per PTR No. 4024-003.
12-19-68	VA1-674	FTP	5.5	1.60	4.98	
12-26-68	VA1-675	25%	15.0	1.60	7.55	
12-27-68	VA1-676	50%	20.0	1.61	5.98	
12-27-68	VA1-677	65%	8.0	1.61	5.58	
12-27-68	VA1-678	65%	10.0	1.62	5.56	
12-27-68	VA1-679	25%	26.0	1.58	7.40	
12-27-68	VA1-680	50%	26.0	1.60	5.81	
1-17-69	VA1-688	25%	40.0	1.60	7.56	New build with all 65 T/C's nickel coated; chamber rotated to new basic angular position.
1-17-69	VA1-689	10%	60.0	1.62	12.15	
1-17-69	VA1-690	65%	10.0	1.61	5.55	
1-17-69	VA1-691	50%	25.0	1.61	5.91	
1-28-69	VA1-692	25%	50.0	1.63	7.90	
1-28-69	VA1-693	FTP	7.7	1.61	4.72	
1-28-69	VA1-694**	25%	40.0	1.59	5.65	

* Approximate Values

** No Barrier Coolant Flow

NOTE: A 1.00 inch injector turbulator was used on all tests.

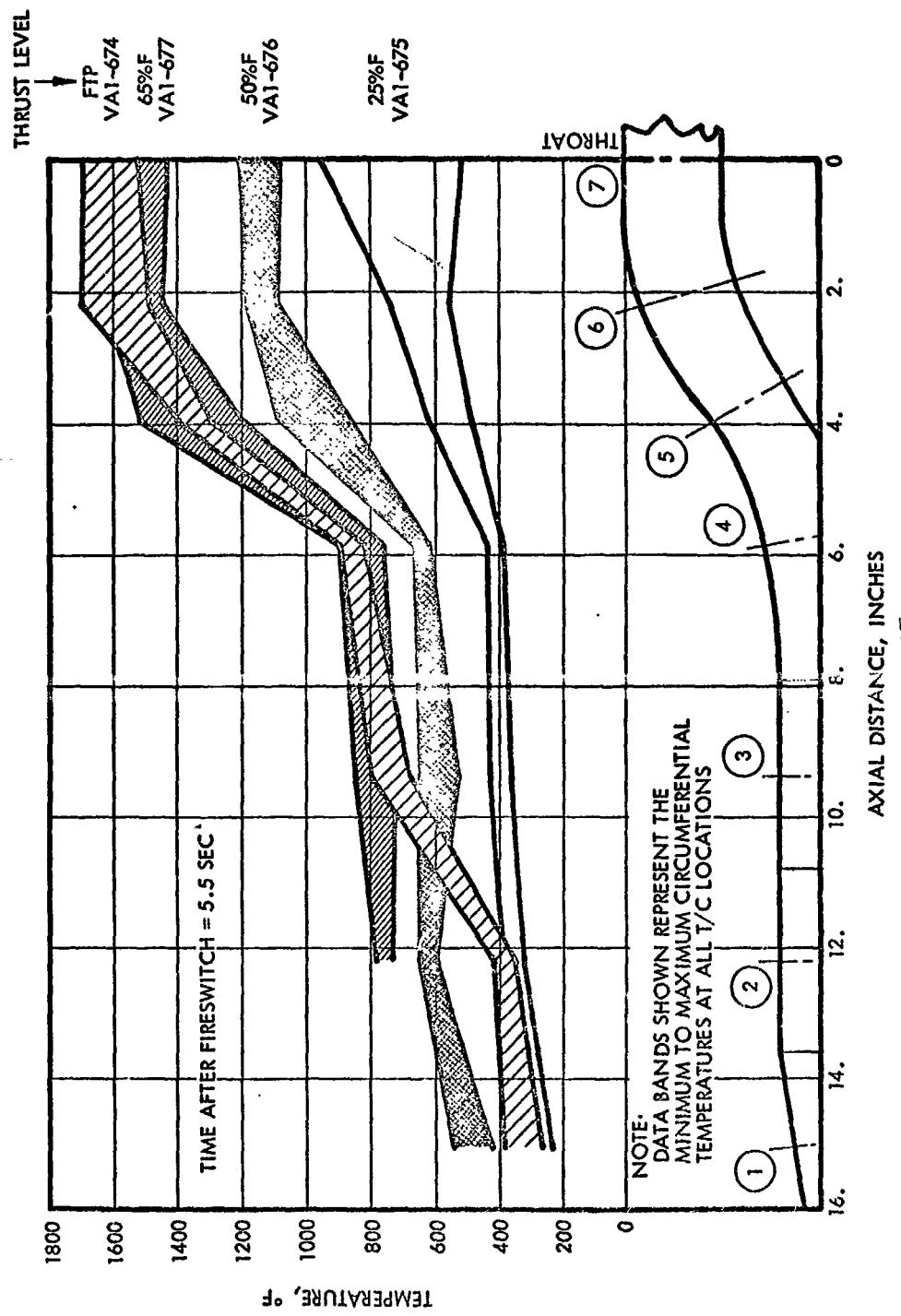


FIGURE 3-9 QUAL B HEAT SINK CHAMBER
MEASURED HOT WALL TEMPERATURE VERSUS AXIAL LOCATION

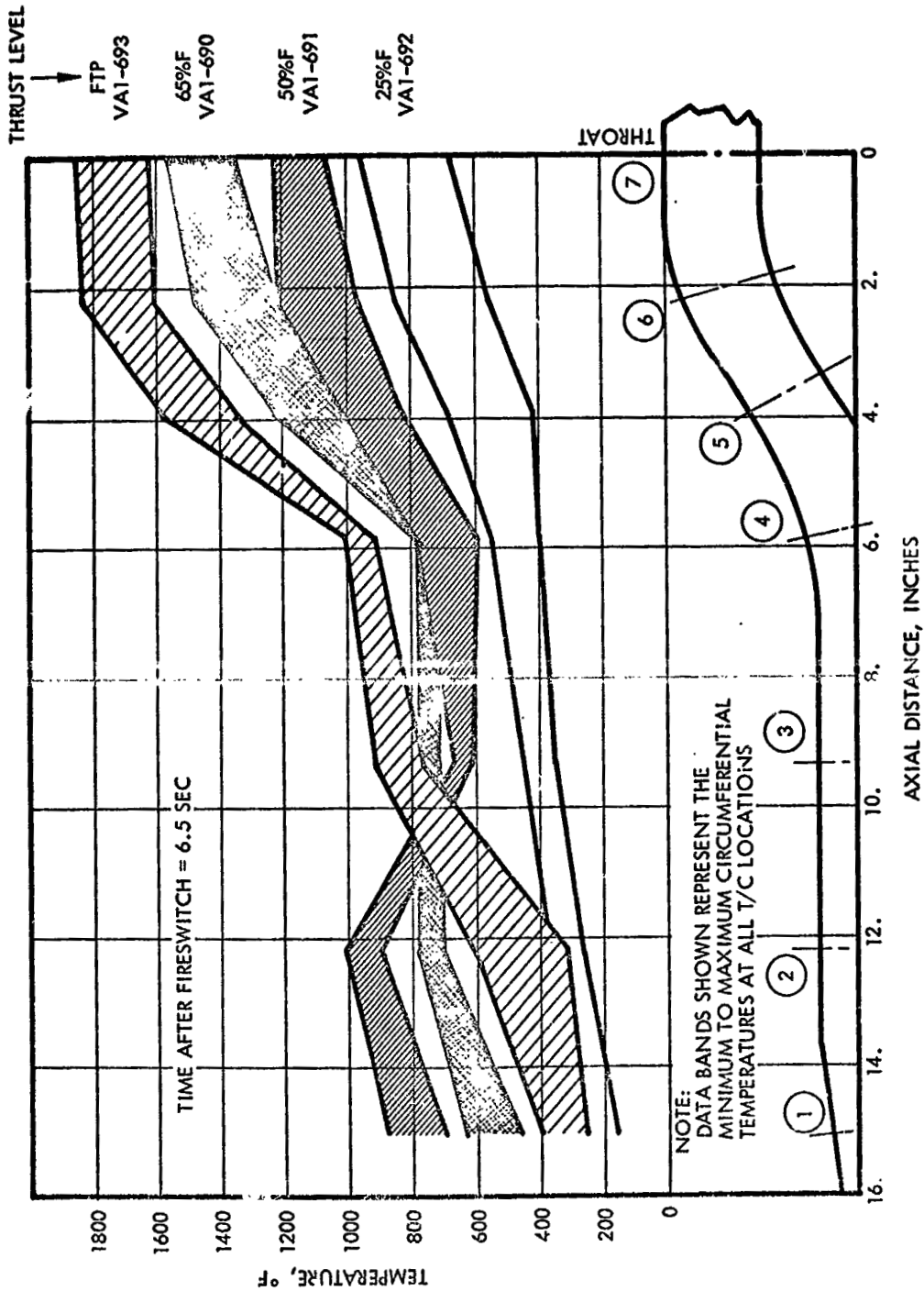


FIGURE 3-10 QUAL 3 HEAT SINK CHAMBER
MEASURED HOT WALL TEMPERATURE VERSUS AXIAL LOCATION

3.5.4.3 Heat Flux Summary

A summary of the throat region heat fluxes for the Qual B steel chamber is presented in Table 3-2.

TABLE 3-2. HEAT SINK HEAT FLUX SUMMARY

Test	% F	T _w , °F	Location	Heat Flux, Btu/sec-in ²		
				Combined ON & OFF B.C.H.*	OFF B.C.H.	ON B.C.H.
VA1-693 ↓	FTP ↓	1000 ↓	4	0.97	0.95	0.98
			5	2.02	1.85	2.18
			6	2.36	2.37	2.36
			7	2.40	2.37	2.44
		1200 ↓	4	0.94	0.91	0.97
			5	1.87	1.73	2.00
			6	2.24	2.24	2.24
			7	2.30	2.20	2.39
		1400 ↓	5	1.65	1.53	1.77
			6	2.12	2.06	2.17
			7	2.16	2.06	2.26
		1600 ↓	6	1.94	1.90	1.98
			7	2.00	1.93	2.06
VA1-690 ↓	65% ↓	1000 ↓	4	0.77	0.77	0.76
			5	1.13	1.11	1.15
			6	1.44	1.47	1.41
			7	1.54	1.53	1.54
		1200 ↓	4	0.76	0.76	0.76
			5	1.11	1.10	1.12
			6	1.44	1.48	1.39
			7	1.60	1.63	1.56

* B.C.H. - Barrier Coolant Hole

Table 3-2 (Continued)

Test	% F	T _w , °F	Location	Heat Flux, Btu/sec-in ²		
				Combined ON & OFF B.C.H.	OFF B.C.H.	ON B.C.H.
VA1-690 (Cont.) ↓	65% ↓	1400 ↓	5	1.08	1.08	1.08
			6	1.40	1.44	1.36
			7	1.53	1.53	1.53
		1600 ↓	6	1.35	1.38	1.31
			7	1.49	1.49	1.49
VA1-691 ↓	50% ↓	1000 ↓	4	0.61	0.60	0.62
			5	0.92	0.88	0.96
			6	1.35	1.47	1.23
			7	1.35	1.47	1.23
		1200 ↓	4	0.60	0.60	0.60
			5	0.82	0.80	0.83
			6	1.13	(1.21)	1.05
			7	1.15	1.16	1.14
		1400 ↓	5	0.77	0.75	0.78
			6	0.92	0.93	0.90
			7	1.04	1.02	1.05
		1600 ↓	6	0.77	0.75	0.78
			7	0.93	0.91	0.95
VA1-692 ↓	25% ↓	1000 ↓	4	0.42	0.415	0.43
			5	0.60	0.55	0.64
			6	0.60	0.60	0.60
			7	0.90	0.86	0.94
		1200 ↓	4	0.42	0.41	0.43
			5	0.58	0.54	0.61
			6	0.59	0.60	0.57
			7	0.79	0.75	0.83
		1400 ↓	5	0.55	0.52	0.58
			6	0.55	0.55	0.54
			7	0.71	0.68	0.74
		1600 ↓	6	0.52	0.52	0.51
			7	0.64	0.61	0.66

Emphasis on heat flux data has been placed on tests VA1-688 through 693 because of the care taken in preparing the thermocouple instrumentation on the heat sink chamber.

Heat flux data at throat locations 6 and 7 have been charted in Figure 3-11 to show the heat flux changes with thrust level for the Qual B H/S configuration.

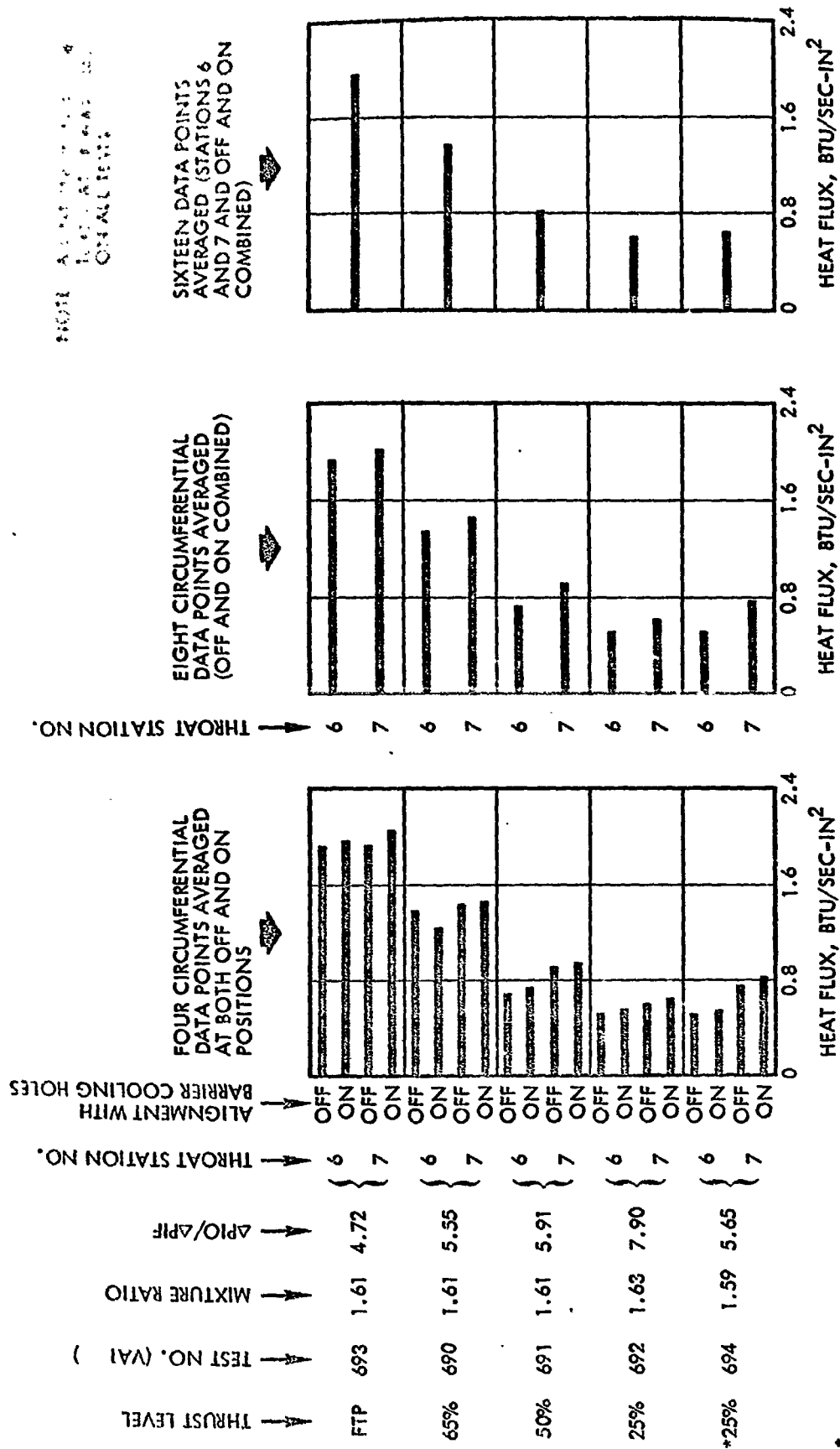
- The throat plane showed the highest incident heat flux when compared with other throat region (convergent) locations for wall temperatures in excess of 1500°F.
- The heat flux at the throat in-line with the barrier cooling holes was (in general) slightly greater than throat points located between barrier cooling holes.
- Statistical analysis and averaging techniques were necessary to adequately define the heat flux levels on and off the barrier cooling holes.
- The heat flux levels in the throat region (locations 6 and 7) were as follows:

Thrust Level	Heat Flux, Btu/sec-in ² (T _w) = 1600°F	
FTP	1.98	(16 Data Point Averages)
65%	1.36	
50%	0.82	
25%	0.58	

3.5.4.4 Indicated Throat Recovery Temperatures

The heat flux data from Table 3-2 was analyzed on a parametric basis revealing the possible combinations of effective local gas recovery temperatures and corresponding film coefficients to satisfy the average heat flux at the throat station for each thrust level investigated.

T_W = 76.8 °F



* NO BARRIER COOLING FLOW

FIGURE 3-11 QUAL B HEAT SINK THROAT HEAT FLUX SUMMARY

Computations were accomplished by use of the following:

$$h_g = \frac{Q/A}{T_R - T_W}$$

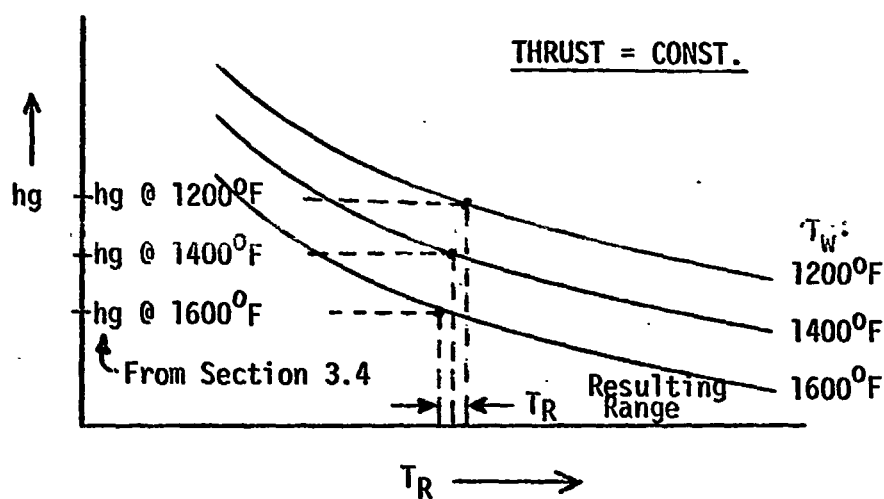
where:

T_W = (Variable as shown)

Q/A - Heat Flux from Table 3-2

T_R - assumed range of values

The following type of curve was then generated to aid in the determination of T_R at each thrust level:



The hot side wall thermal environment did not exhibit constant recovery temperature/heat transfer film coefficient levels during heat sink tests with wall temperatures up to 1800°F.

The indicated recovery temperatures at the throat plane (location 7) were as shown on the following page.

Thrust Level	Recovery Temperature, °F	Comments
FTP	4600.	(Evaluated from data with a wall temperature range of 1200°F to 1600°F)
65%	4650.	
50%	4070.	
25%	4300.	

These results are based on the assumption that nominal theoretical film coefficients from Section 3.4 exist at the throat. The recovery temperatures shown are, at best, approximate values. The use of theoretical film coefficients grants the only means possible (at the present time) to derive recovery temperature information from the heat flux data. The results of this study are shown graphically in Figure 3-12.

3.5.5 Modified Chamber Contour Thermal Environment Characterization

The modified chamber contour test series was initiated on 18 February 1969 and completed on 21 March 1969. Testing was conducted on HEA 023 in the VETS facility. The total reactive time accumulated was 480 seconds consisting of 8 Countour "A" tests, 5 Contour "B" tests, and 8 Contour "C" tests. The thermal environment information has been compared with data from the Qual B throat configuration to yield the contour changes most likely to result in reducing LMDE throat heat flux and erosion.

3.5.5.1 Modified Contour Test Summary

The modified chamber contour tests utilizing a common conical instrumented heat sink throat are summarized in Table 3-3.

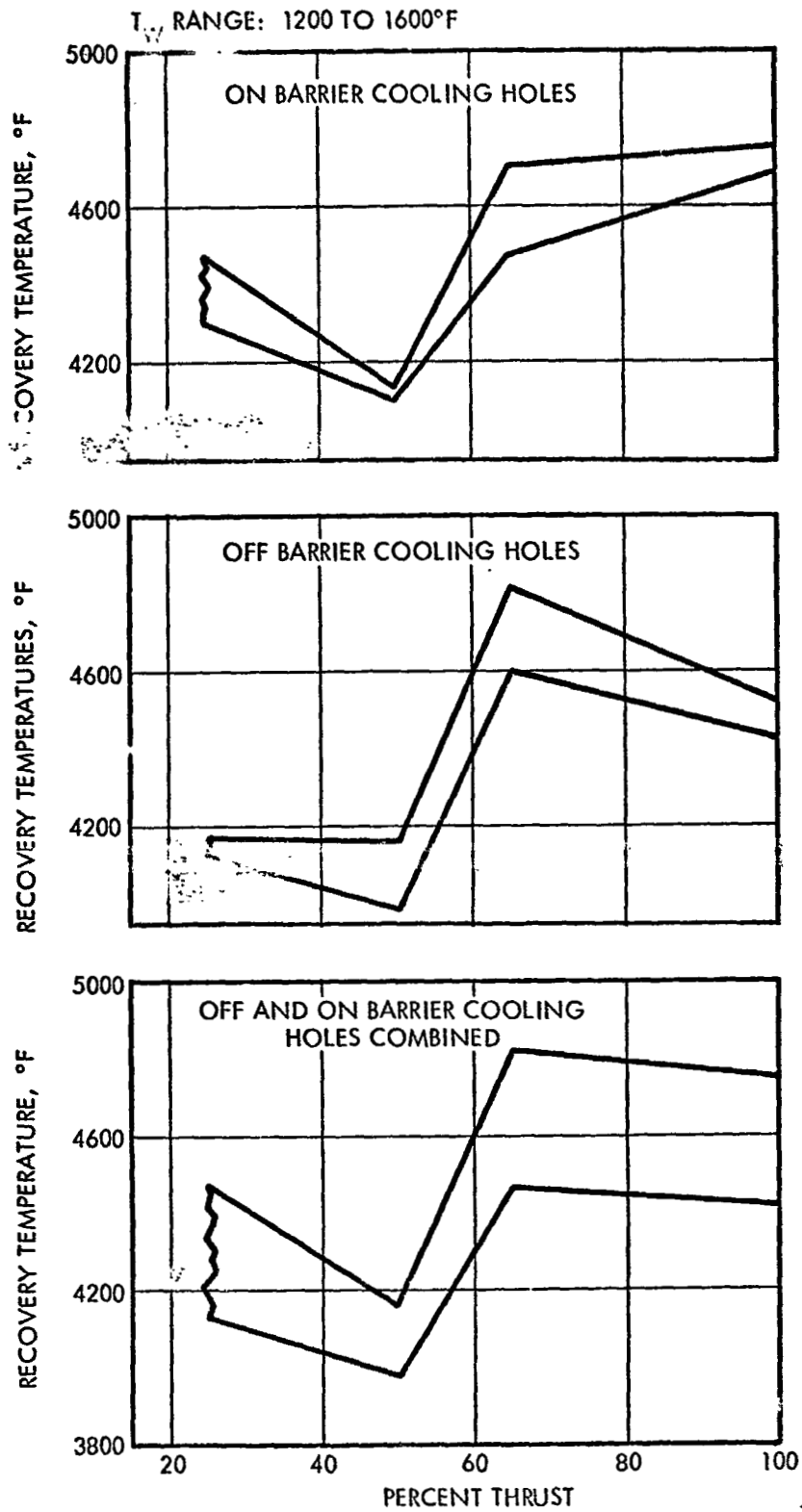


FIGURE 3-12 QUAL B THROAT STATION 7 ESTIMATED RECOVERY TEMPERATURES

TABLE 3-3. MODIFIED CHAMBER CONTOUR TESTS

Contour	Date	Test No.	Thrust	Duration* (Secs.)	Mixture* Ratio	$\frac{\Delta P_{IO}}{\Delta P_{IF}}$	Comments
"A"	2-18-69	VA1-695	FTP	7.2	1.62	4.78	New build with 37 nickel-coated T/C's in throat; Fiberglass chamber wall.
"A"	2-18-69	VA1-696	FTP	13.0	1.59	4.73	
"A"	2-18-69	VA1-697	25%	56.0	1.68	8.03	
"A"	2-18-69	VA1-698	25%	47.0	1.60	7.23	
"A"	3-11-69	VA1-699	FTP	11.5	1.60	4.78	New build with 37 nickel-coated T/C's in throat; steel chamber wall.
"A"	3-11-69	VA1-700	25%	52.0	1.59	6.90	
"A"	3-11-69	VA1-701	50%	25.0	1.61	5.75	
"A"	3-11-69	VA1-702	65%	15.0	1.61	5.52	
"B"	3-18-69	VA1-704	FTP	13.0	1.59	4.74	New build with 37 nickel-coated T/C's in throat; hard ablative chamber wall.
"B"	3-18-69	VA1-705	25%	50.0	1.59	7.11	
"B"	3-18-69	VA1-706	50%	25.0	1.61	5.98	
"B"	3-18-69	VA1-707	65%	14.0	1.61	5.58	
"B"	3-18-69	VA1-708	FTP	10.0	1.61	4.78	
"C"	3-19-69	VA1-709	FTP	9.0	1.60	4.81	New build with 37 nickel-coated T/C's in throat; hard ablative chamber wall.
"C"	3-20-69	VA1-710	FTP	6.4	1.58	4.73	
"C"	3-20-69	VA1-711	25%	50.0	1.57	7.09	
"C"	3-20-69	VA1-712	50%	25.0	1.60	5.90	
"C"	3-20-69	VA1-713	65%	12.0	1.60	5.50	
"C"	3-21-69	VA1-714*	FTP	9.0	1.54	4.90	
"C"	3-21-69	VA1-715**	FTP	8.0	1.60	5.17	
"C"	3-21-69	VA1-716***	FTP	6.5	1.59	5.31	

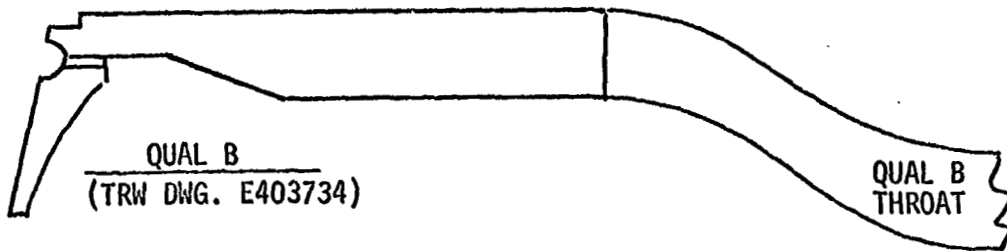
* Approximate values.

** 16% barrier coolant flow (barrier coolant orifice removed).

*** 1.00 inch diameter injector turbulator removed; a 0.96 inch diameter turbulator installed.

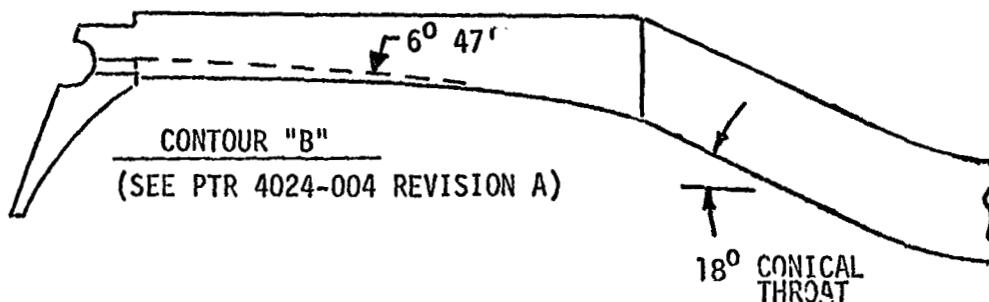
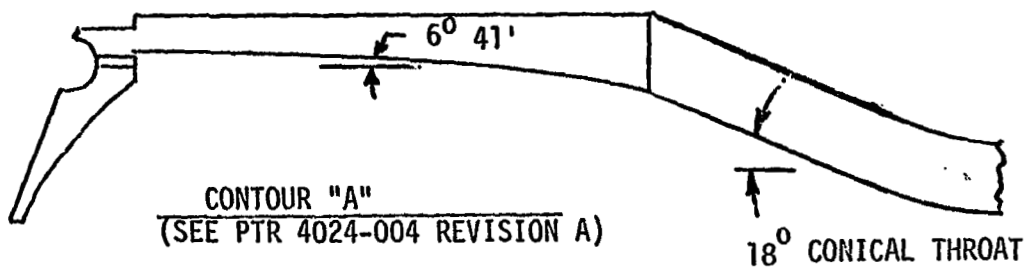
3.5.5.2 Experimental Chamber Configurations

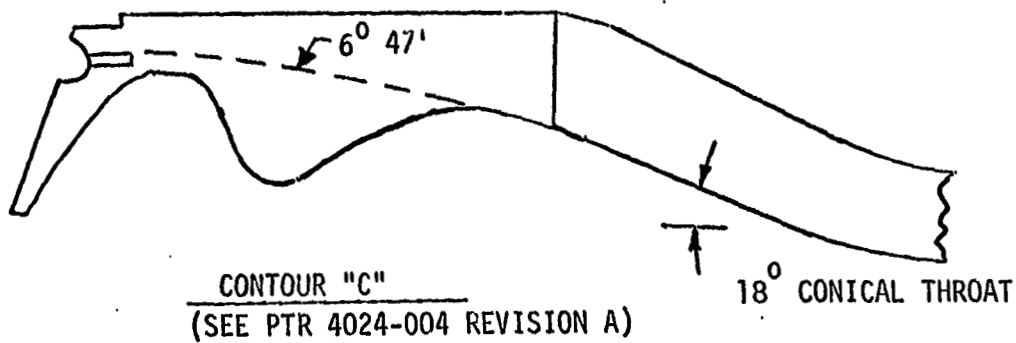
The heat flux data from all tests were obtained from the following thrust chamber configurations:



(Illustrations are not to scale)

The Qual B configuration above has been analyzed and the results of the study have been documented in Section 3.5.4 as the baseline thermal characterization.





A common heat sink conical throat was used on all tests of the modified chamber contours. This throat (Figure 3-13 below) was used to obtain transient hot wall temperature data by use of thermocouple probes located at the hot side surface.

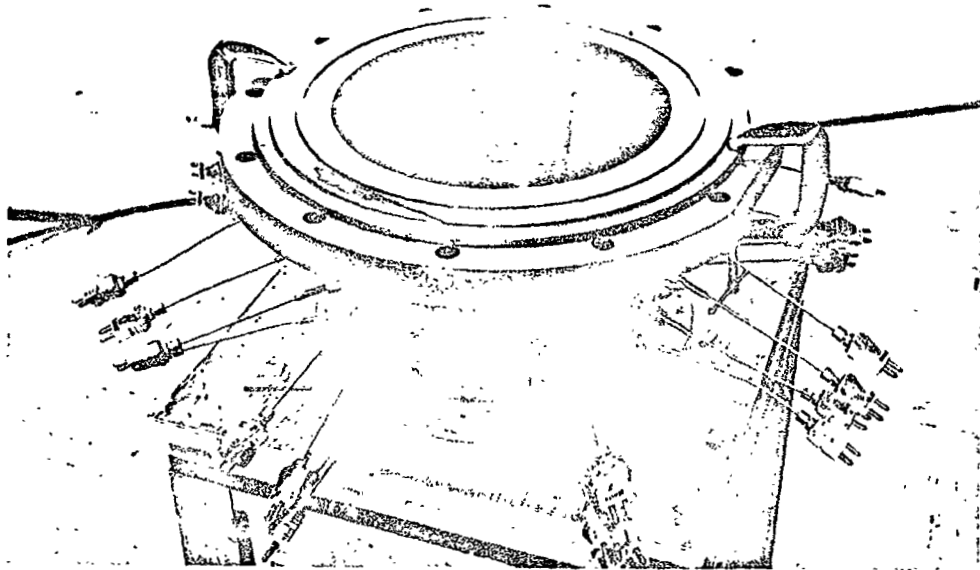


FIGURE 3-13 HEAT SINK CONICAL THROAT PHOTOGRAPH

These temperature data were used to calculate the throat region heat flux levels by use of Section 3.5.3 data reduction program.

Typical transient temperature traces from strip charts taken during the tests are presented in Reference 26.

3.5.5.3 Heat Flux Summary and Comparisons with Qual B

The thermal data acquired from these tests has produced the following information:

- Each modified chamber configuration yielded lower throat region heat loads than the Qual B configuration at thrust levels of 25%, 65%, and FTP.
- The 50% thrust level tests showed that the modified chambers yielded higher throat heat loads than the Qual B configuration.
- The heat flux changes due to the modified contours relative to the Qual B baseline throat heat flux level are as follows:

Thrust Level	Q/A Modified Contour			
	Q/A Qual B (Btu/sec-in ²)	Q/A Qual B		
		"A" (d'less)	"B" (d'less)	"C" (d'less)
FTP	1.98	0.70	0.76	0.90
65%	1.36	0.98	0.96	0.97
50%	0.82	1.06	1.13	1.09
25%	0.58	0.90	0.83	0.55

NOTE: ● Q/A values from data at $T_w = 1600^\circ\text{F}$
 ● Q/A values are sixteen data point averages for throat stations 6 and 7

- The modified chamber contour tests (utilizing a conical throat) showed that the heat flux at station 6 upstream of the throat plane exhibited the highest heat flux in the throat region.
- The 65% thrust throat heat flux was relatively insensitive to the modified chamber contour changes.

- Modified chamber contour "C" produced a 45 percent reduction in throat heat load at 25% thrust when compared with the Qual B configuration. All other thrust levels showed contour "C" to exhibit very little ($\leq 10\%$ reduction) heat flux changes compared with Qual B levels.

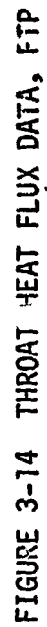
Bar charts of the heat flux information are presented in Figures 3-14 through 3-21. All modified chamber contour throat heat flux levels are presented at fixed thrust levels in Figures 3-14 through 3-17 on a contour comparison basis. In addition, throat heat flux levels are presented for each individual contour in Figures 3-18 through 3-21 on a thrust level comparison basis showing the heat flux reduction with reduced thrust.

3.5.5.4 Improved Flow Distribution Test Series

The experimental heat transfer studies for the improved flow distribution test series were conducted using a head end assembly fitted with the XX2 injector (as opposed to the Qual HEA 023 used in all previous testing). All of the tests performed used either modified chamber Contours "A" or "C". The following sections present the chamber heat flux levels resulting from the tests in which the flow distribution is improved by varying the oxidizer turbulator diameter. These results are presented and compared for a range of turbulator sizes for Contour "A" and two sizes for Contour "C". These comparisons aid in the selection of a turbulator size most likely to result in reduced LMDE throat erosion.

Test Summary

The improved flow distribution test series was initiated on 12 May 1969 and completed on 5 June 1969. Testing was conducted on HEA XX2 in the VETS facility. The test series is defined in Reference 27 and summarized in Table 3-4.



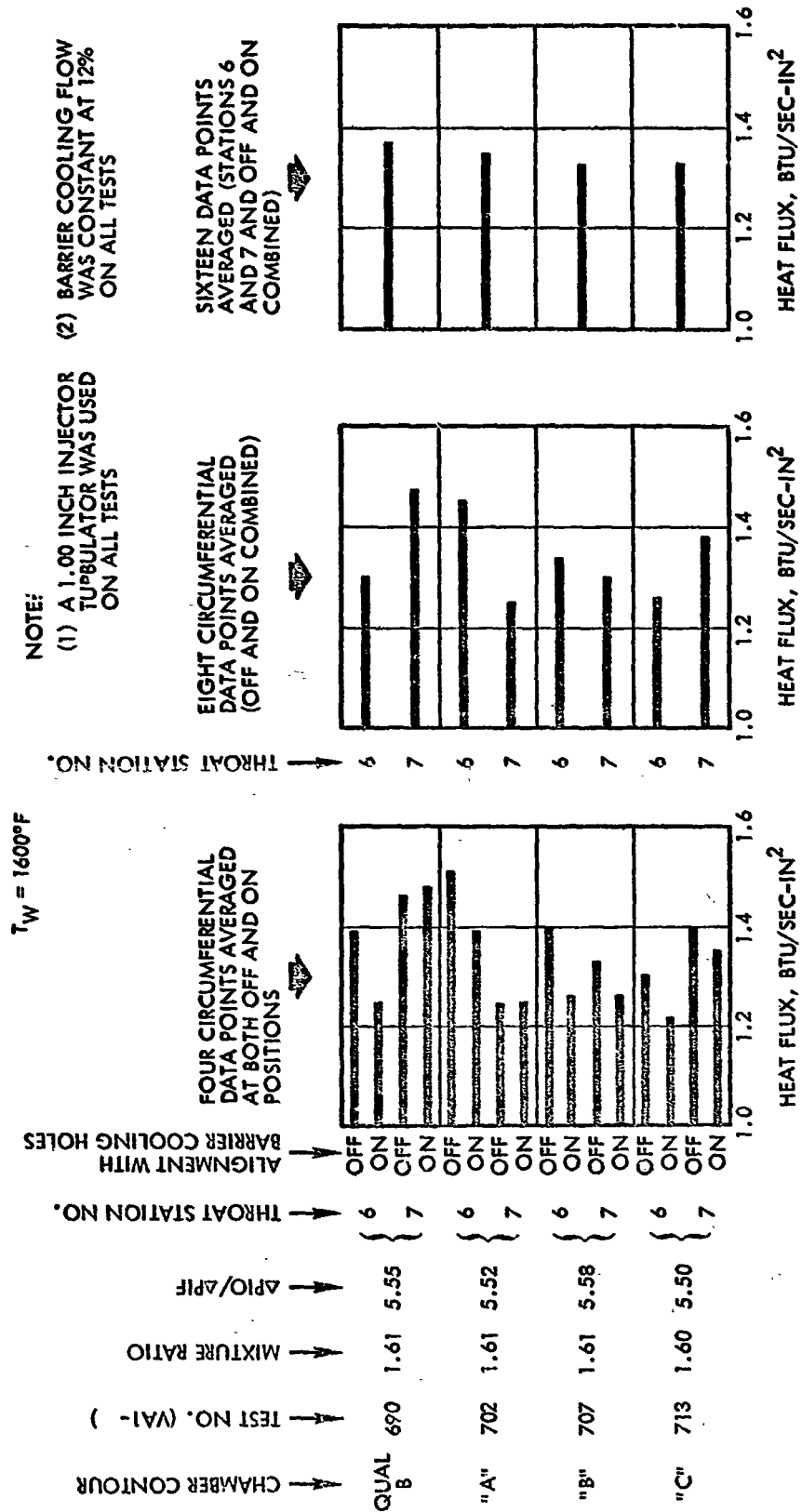


FIGURE 3-15 THROAT HEAT FLUX DATA, 65% THRUST

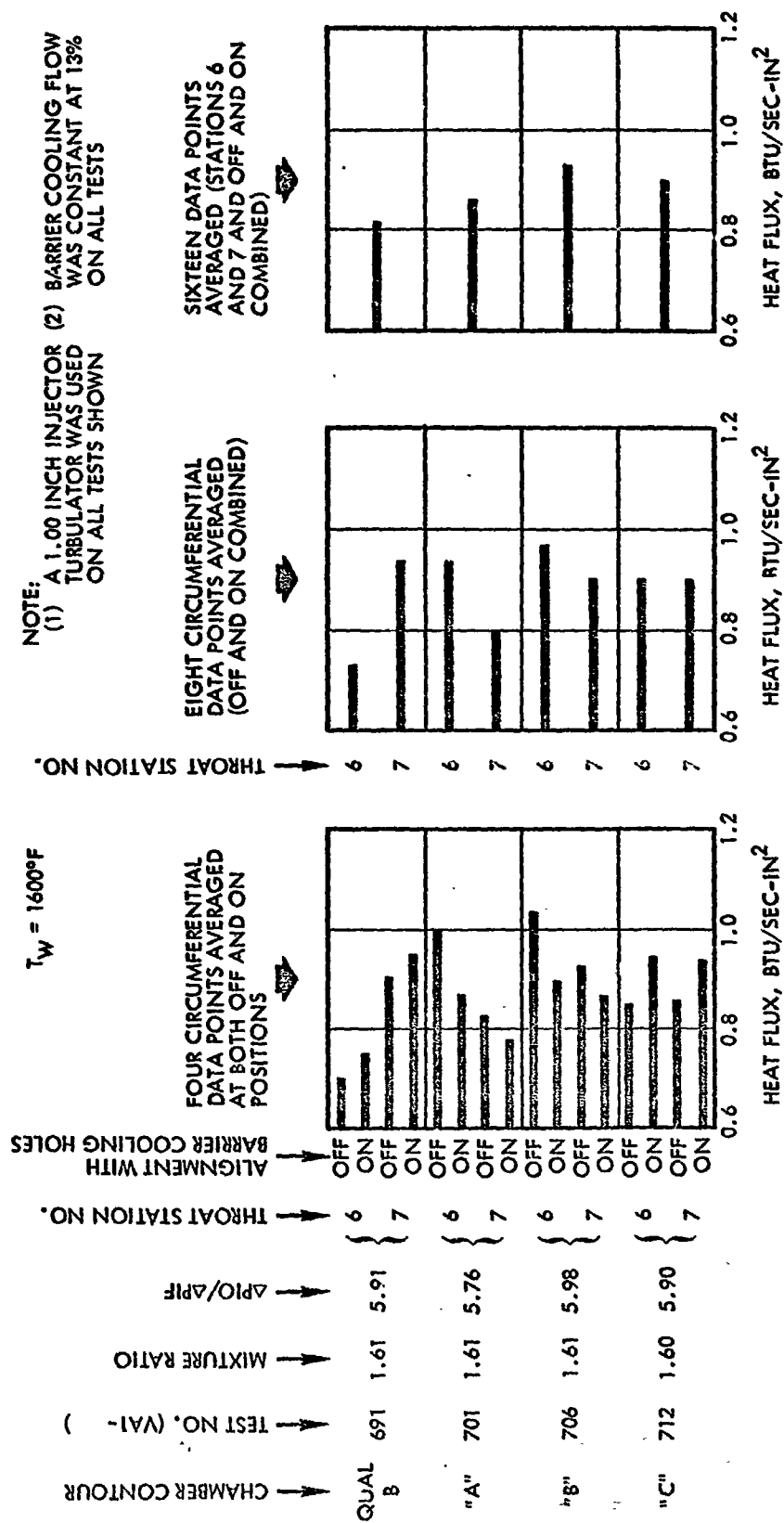


FIGURE 3-16 THROAT HEAT FLUX DATA, 50% THRUST

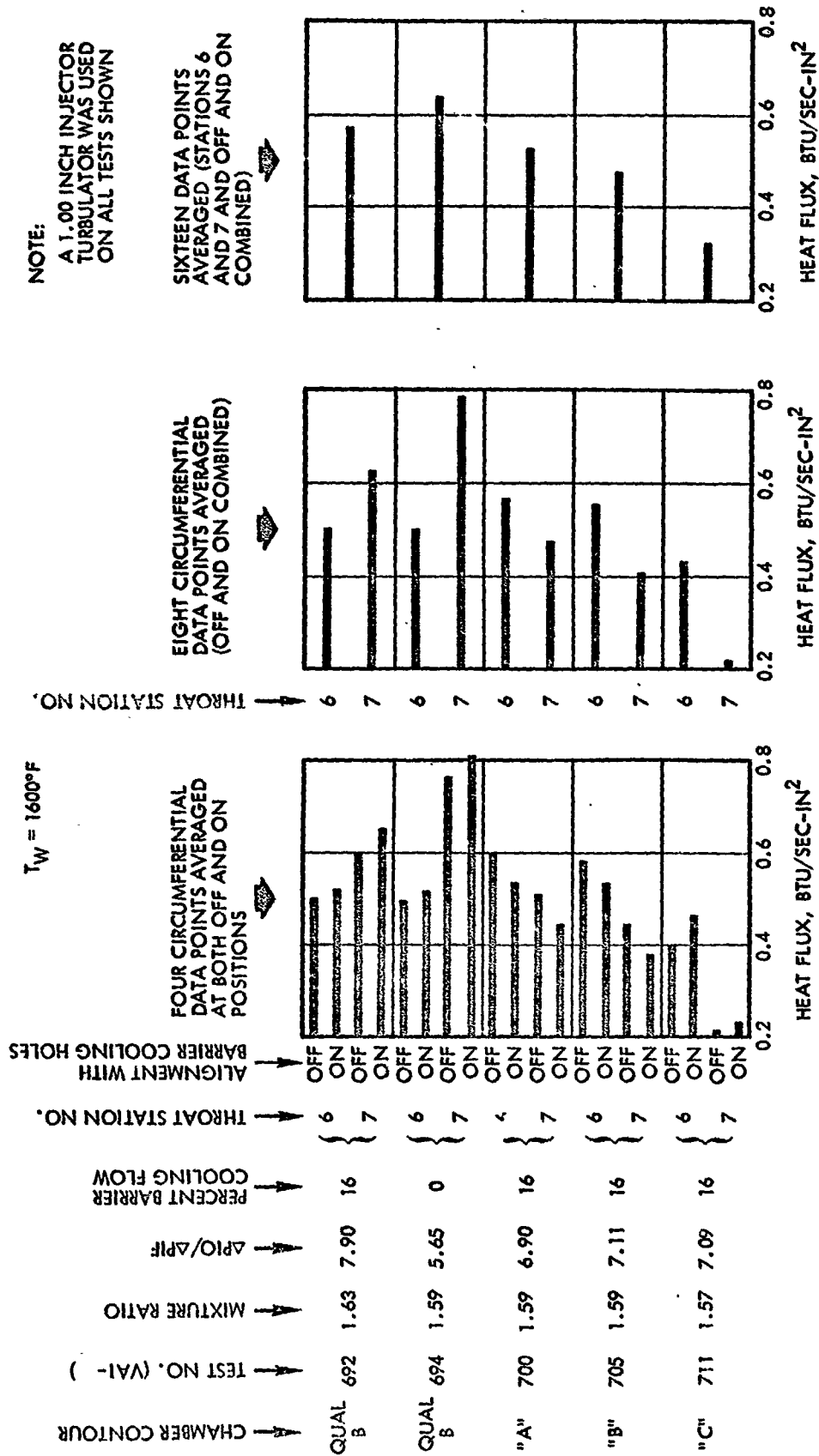


FIGURE 3-17 THROAT HEAT FLUX DATA, 25% THRUST

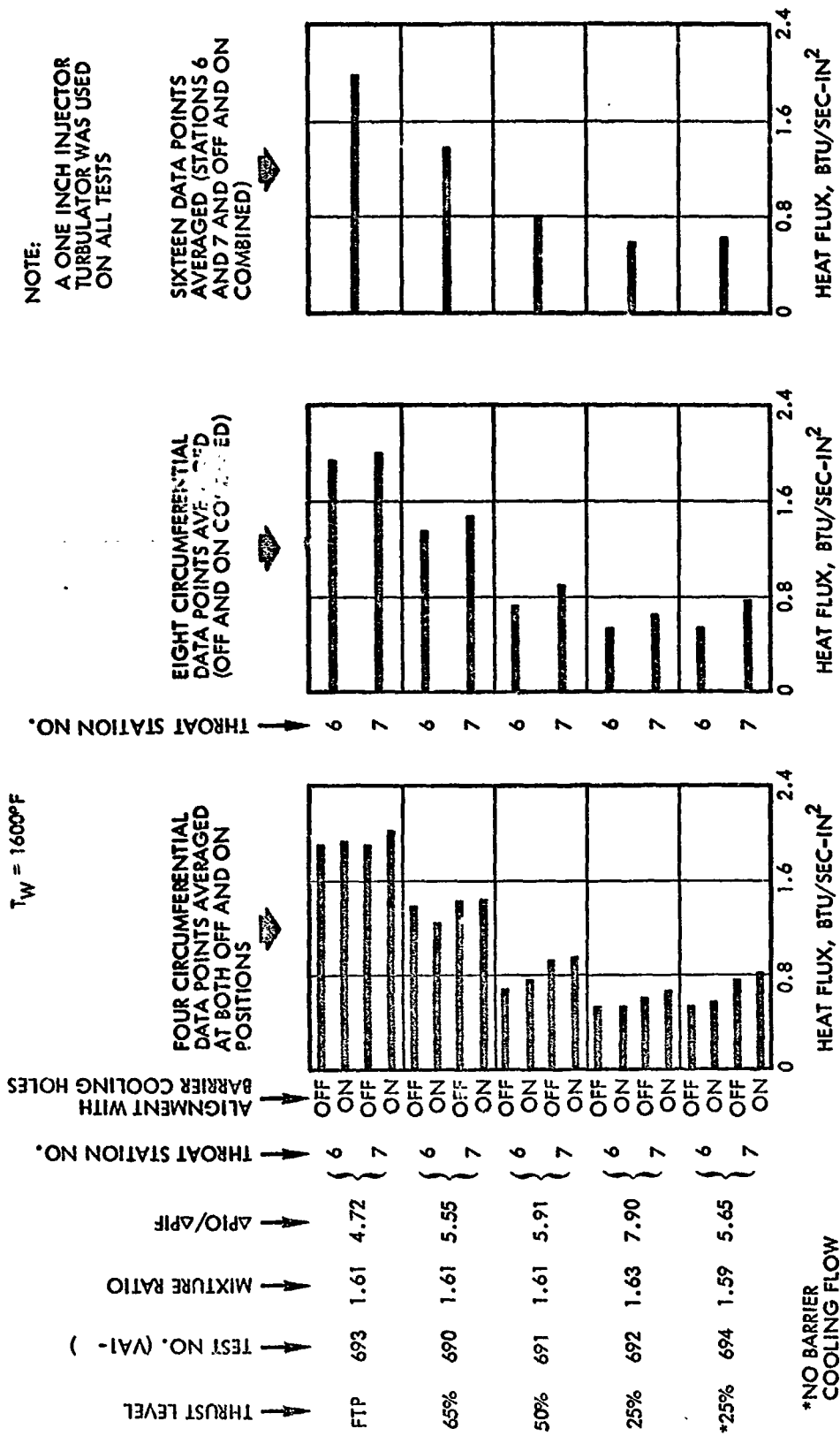


FIGURE 3-18 THROAT HEAT FLUX DATA, QUAL B CHAMBER CONFIGURATION

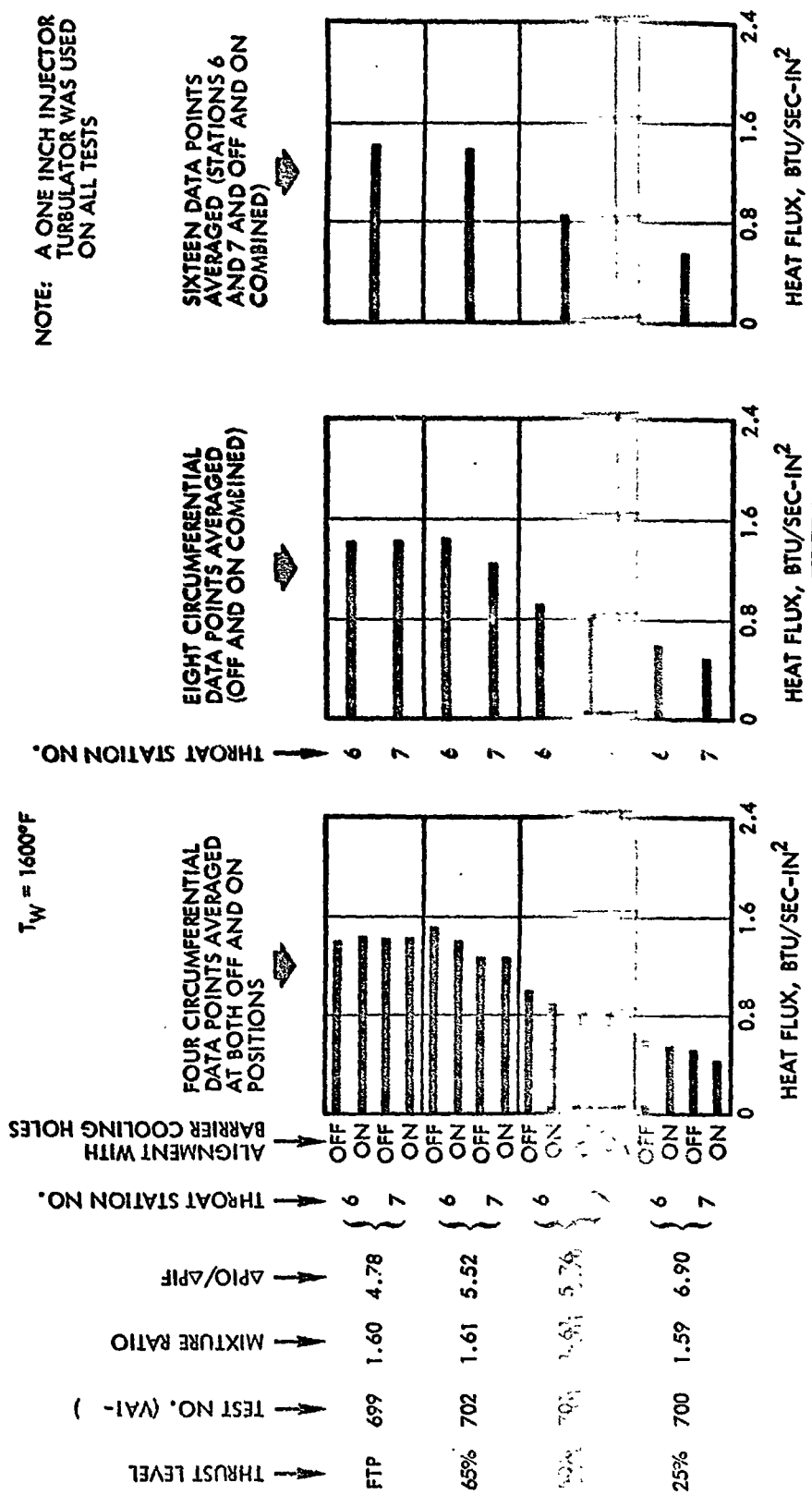


FIGURE 3-19 THROAT HEAT FLUX DATA, MULTIPLIED CHAMBER CONTOUR "A"

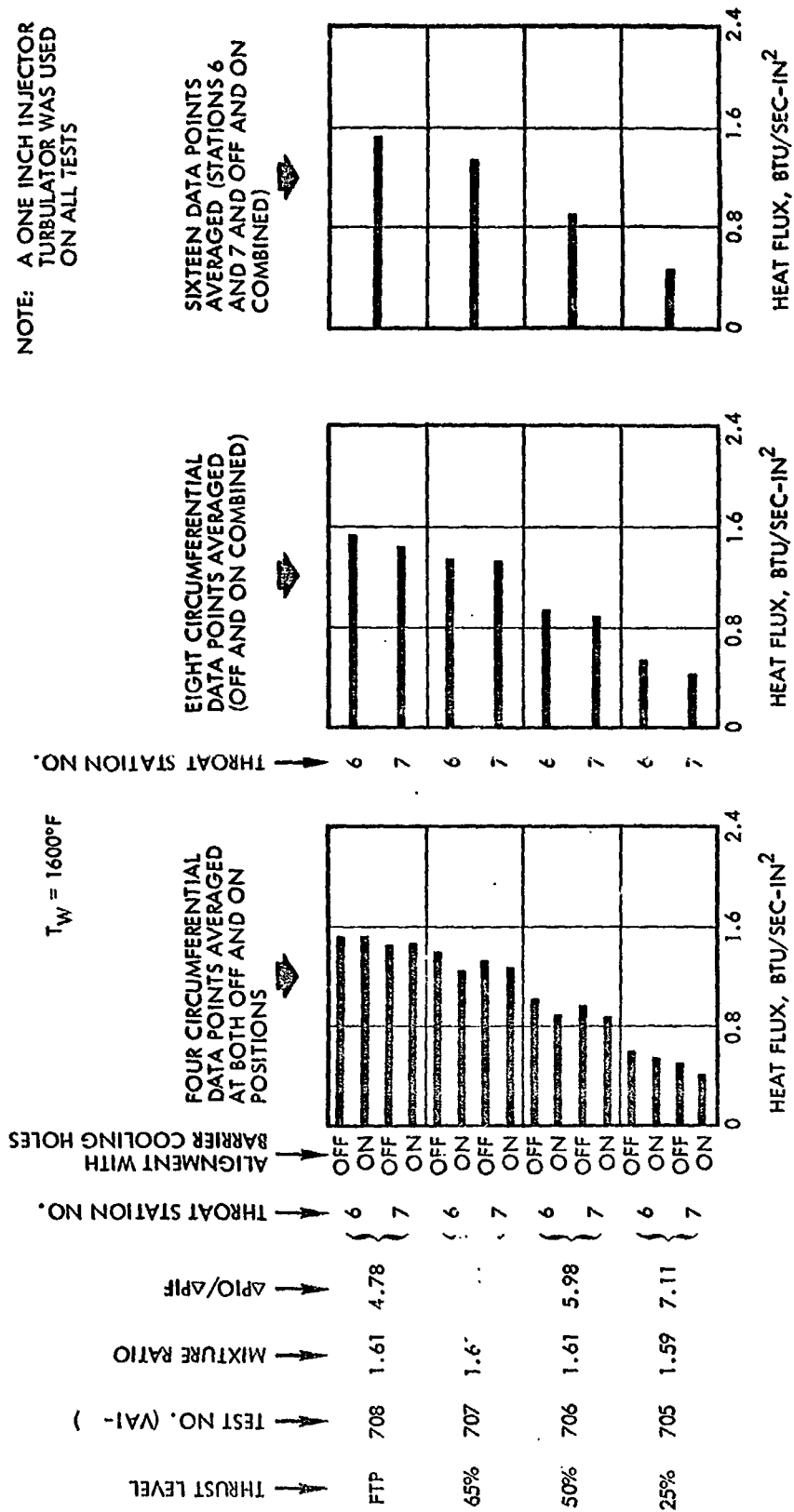


FIGURE 3-20 THROAT HEAT FLUX DATA, MODIFIED CHAMBER CONTOUR "B"

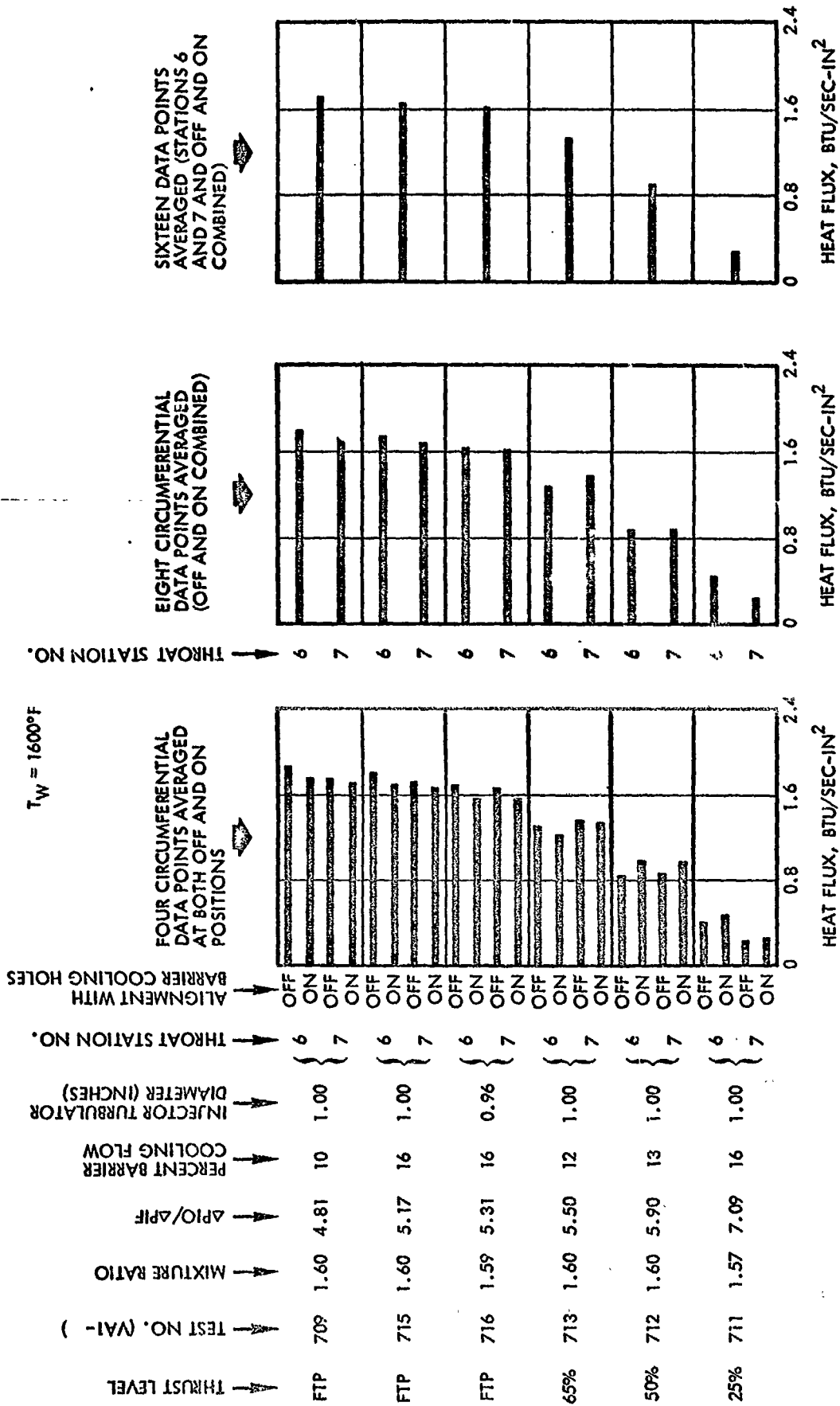


FIGURE 3-21 THROAT HEAT FLUX DATA, MODIFIED CHAMBER CONTOUR "C"

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TABLE 3-4. IMPROVED FLOW DISTRIBUTION TEST SERIES

Date	Test No.	Contour	Thrust	Duration (sec)	Mixture Ratio	Turbulator Size (in)
5-23-69	VA2-644	"A"	FTP	11.	1.60	1.025
5-23-69	VA2-645	"A"	65%	18.	1.64	1.025
5-23-69	VA2-646	"A"	65%	14.	1.58	1.025
5-23-69	VA2-647	"A"	65%	12.	1.60	1.025
5-23-69	VA2-648	"A"	FTP	11.	1.60	1.000
5-23-69	VA2-649	"A"	FTP	13.	1.60	0.928
5-23-69	VA2-650	"A"	65%	17.	1.54	0.928
5-26-69	VA2-651	"A"	65%	13.	1.60	0.928
5-26-69	VA2-652	"A"	FTP	14.	1.60	0.960
6-10-69	VA2-653	"C"	FTP	10.	1.58	0.928
6-10-69	VA2-654	"C"	FTP	9.	1.60	0.928
6-10-69	VA2-655	"C"	FTP	8.	1.60	1.000

Heat Flux Summary and Comparisons

A common heat sink conical throat (Figure 3-13) was used to obtain transient hot wall temperature data by use of thermocouple probes located at four axial locations at the hot side surface. Stations 4, 5, and 6 are located at the entrance to the throat and Station 7 is located at the throat plane.

Heat flux levels are obtained from the temperature data by means of the data reduction program described in Section 3.5.3. The data acquired from these tests has produced the following information:

- For chamber Contour "A" at FTP, the 0.928 turbulator yielded the lowest heat flux of all turbulators tested.

- Over the range of wall temperatures experienced during the tests, the 0.960 and 1.025 turbulators yielded heat fluxes lower than that of the standard 1.000 turbulator, with the 1.025 turbulator being the lowest of the three.
- Extrapolation of the heat flux data to temperatures several hundred degrees above those experienced in the tests showed that heat fluxes with the 0.960 turbulator would be lower than that with the 1.025 turbulator. The extrapolation also indicated that at still higher wall temperatures (i.e., close to actual operating conditions) the 1.025 turbulator would yield the maximum heat flux.
- For Contour "A", at 65%, the 0.928 turbulator showed lower heat fluxes than the 1.025" turbulator in the region in the immediate vicinity of the throat plane, but slightly higher heat fluxes in the rest of the throat. Extrapolation of the data to higher wall temperatures shows that the upstream heat flux decreases with the smaller turbulator size.
- For Contour "C", at FTP, the 0.928 turbulator yielded lower heat fluxes at all stations than the 1.000 turbulator.
- The wall heat fluxes at the various stations in the throat region at which measurements were made are as follows:

Contour	Thrust Level	Turbulator Size (inches in diameter)	Q/A (Btu/in ² -sec) at 1600°F				
			Station				Average of 6 and 7
			4	5	6	7	
"A"	FTP	1.025	1.11	1.30	1.59	1.48	1.55
"A"	FTP	1.000	1.21	1.52	1.83	1.63	1.73
"A"	FTP	0.960	1.21	1.54	1.70	1.56	1.63
"A"	FTP	0.928	1.03	1.30	1.43	1.37	1.40
"A"	65%	1.025	1.06	1.30	1.68	1.40	1.54
"A"	65%	0.928	1.08	1.39	1.57	1.25	1.41
"C"	FTP	1.000	1.18	1.38	1.85	1.75	1.80
"C"	FTP	0.928	1.03	1.21	1.51	1.54	1.53

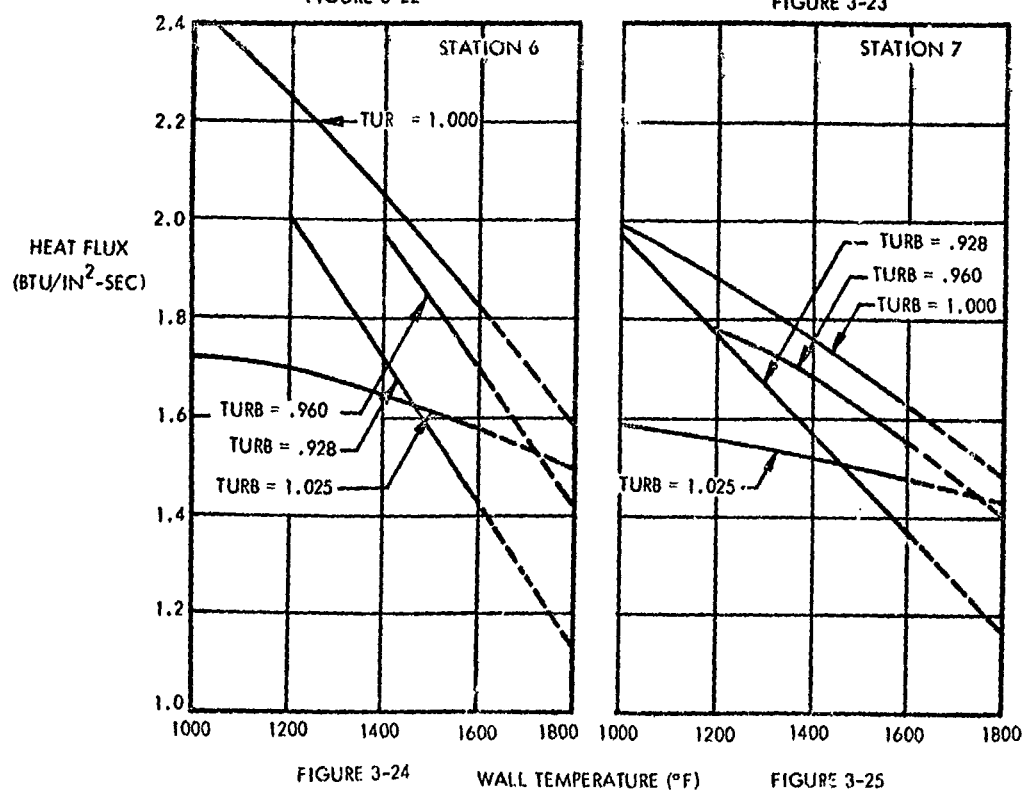
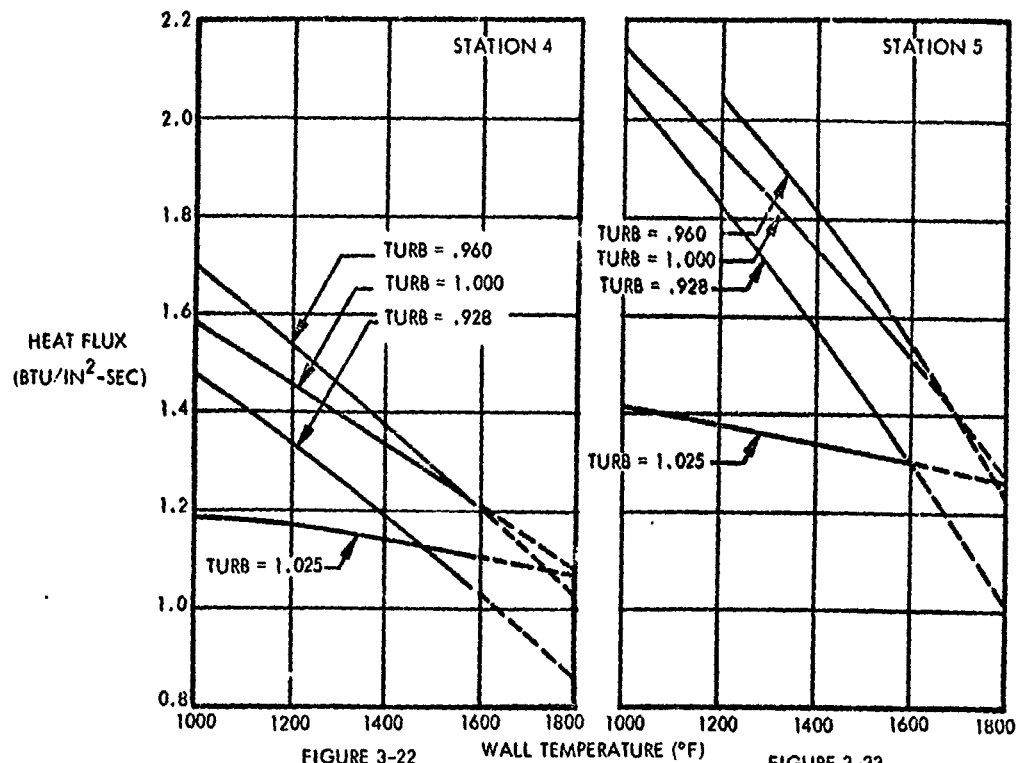
- The reduction in heat flux at FTP in the region in the immediate vicinity of the throat plane which was obtained using a 0.928 turbulator instead of the standard 1.000 turbulator is presented below for each contour tested.

Contour	Station	1.000 Turb.	0.928 Turb.	
		Q/A (Btu/in ² -sec)	Q/A (Btu/in ² -sec)	Percent Reduction
"A"	6	1.83	1.43	-21.9
"A"	7	1.63	1.37	-15.9
"A"	Average of 6 and 7	1.73	1.40	-19.1
"C"	6	1.85	1.51	-18.4
"C"	7	1.75	1.54	-12.0
"C"	Average of 6 and 7	1.80	1.53	-15.0

Above Q/A values have been evaluated at a wall temperature of 1600°F.

- The reduction in heat flux in the region in the immediate vicinity of the throat plane of Contour "A" obtained by using a 0.928 turbulator instead of a 1.025 turbulator is presented on page 3-52 for each contour tested.

At each axial location the heat flux to the walls is determined by taking the average of eight equally spaced circumferential points, four of them being on the barrier coolant holes and four of them off the barrier coolant holes. Figures 3-22 through 3-25 present the average heat flux at each station as a function of wall temperature for each turbulator size tested for Contour "A" at FTP. Figure 3-26 shows data for the average heat flux for Stations 6 and 7. Figures 3-27 through 3-31 present the same information for Contour "A" at 65%. Figures 3-32 through 3-36 present the same information for Contour "C" at FTP. Most of the test data was for a wall temperature of less than 1600°F but extrapolation has been conducted up to 1800°F as shown.



CONTOUR "A" HEAT FLUX LEVELS AT FTP

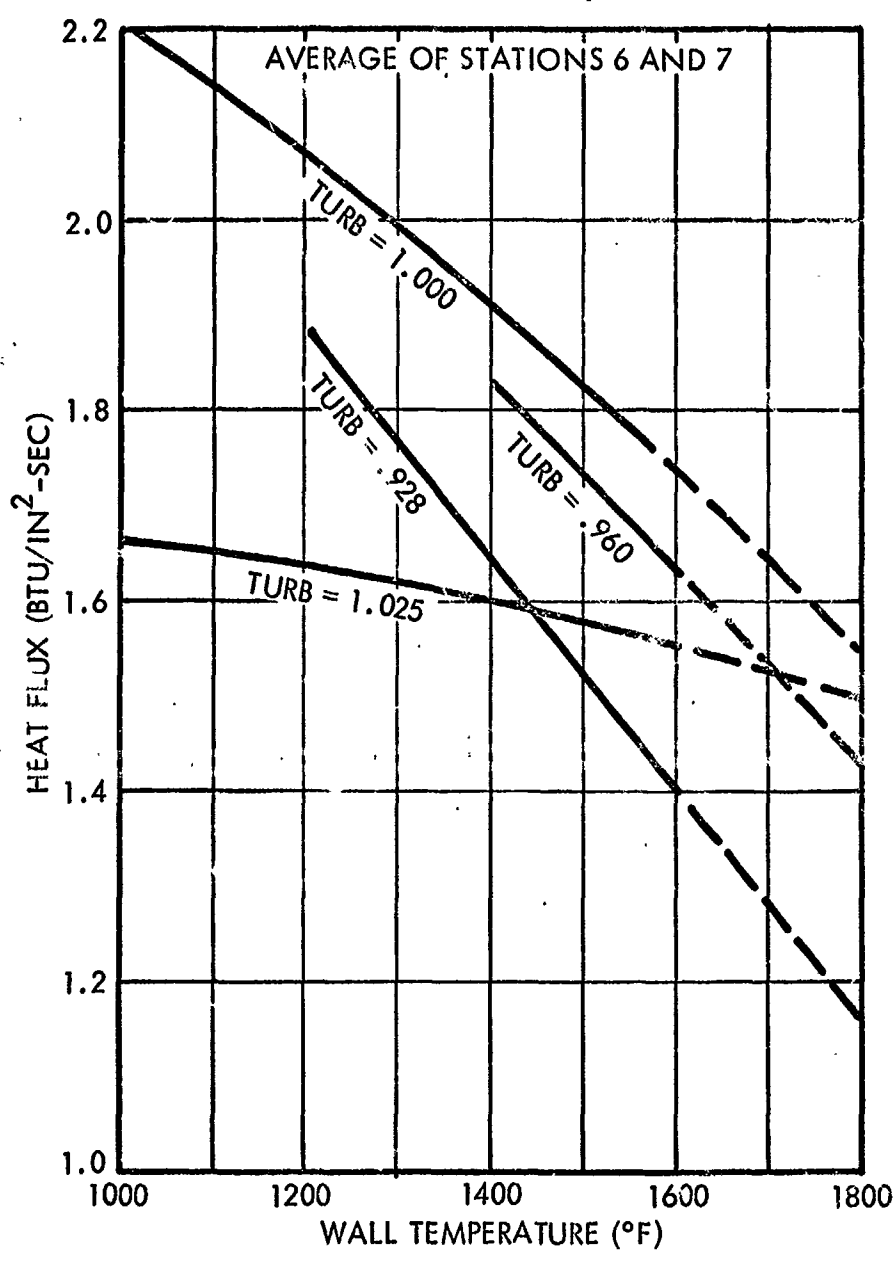


FIGURE 3-26 CONTOUR "A" HEAT FLUX LEVELS AT FTP

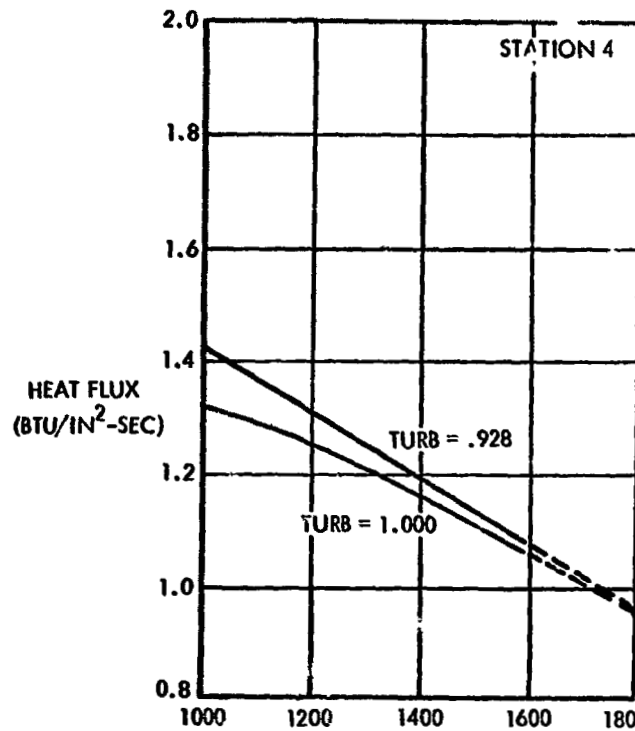


FIGURE 3-27

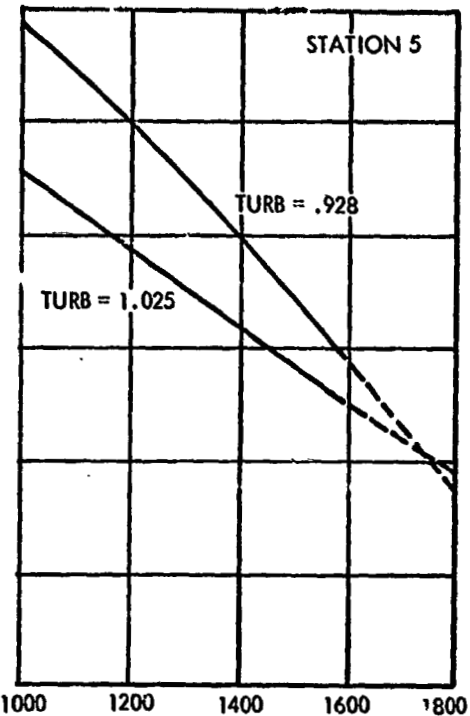


FIGURE 3-28

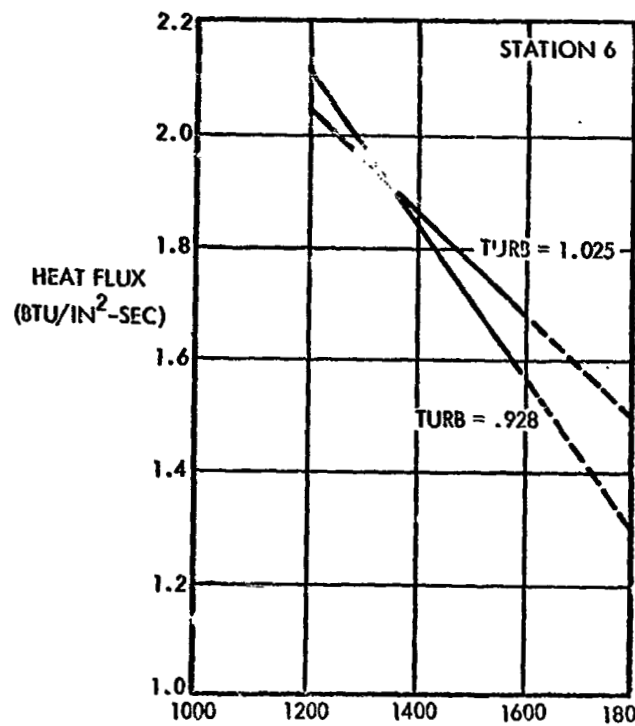


FIGURE 3-29

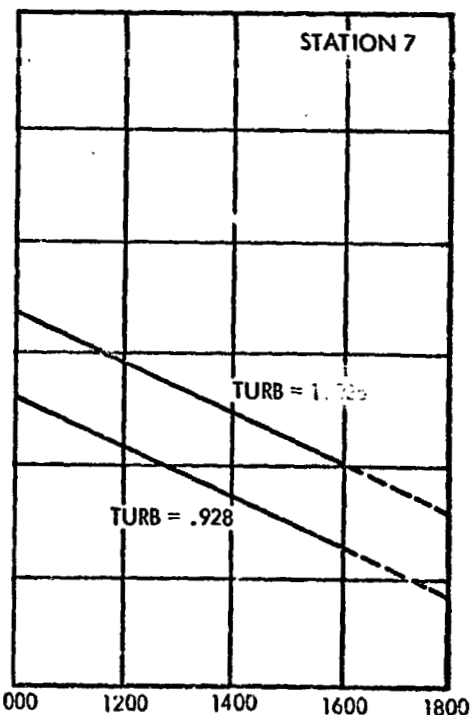


FIGURE 3-30

CONTOUR "A" HEAT FLUX LEVELS AT 65% THRUST

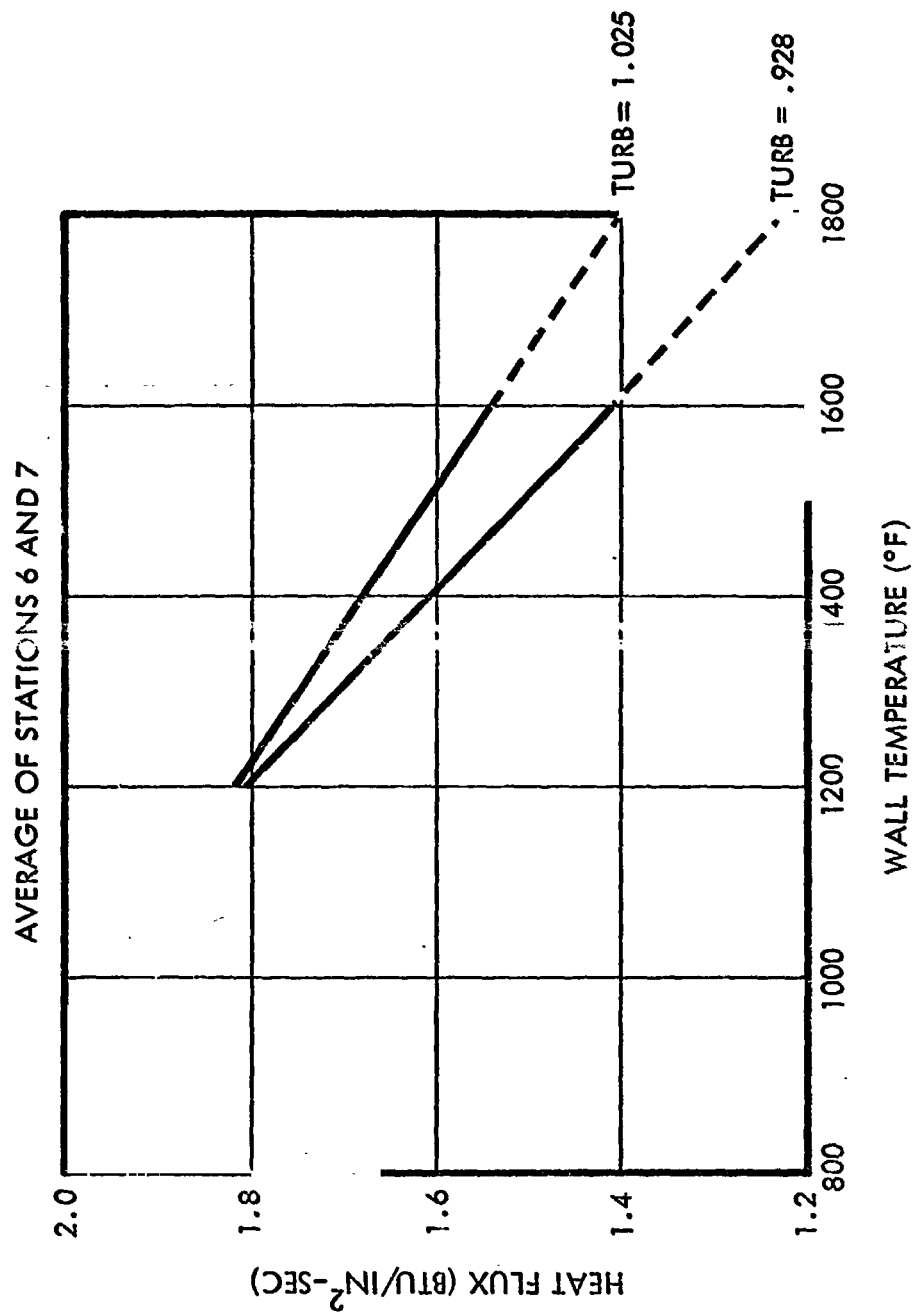


FIGURE 3-31 CONTOUR "A" HEAT FLUX LEVELS AT 65% THRUST

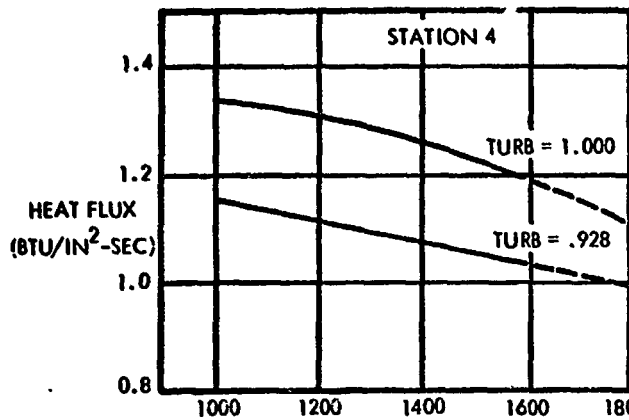


FIGURE 3-32

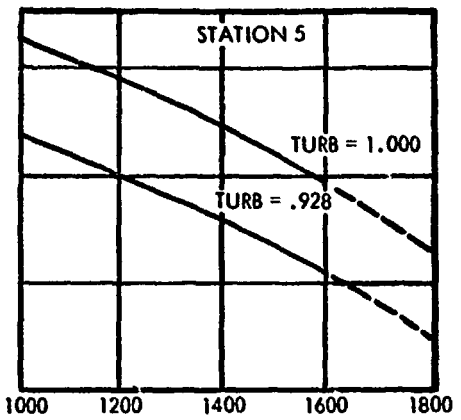


FIGURE 3-33

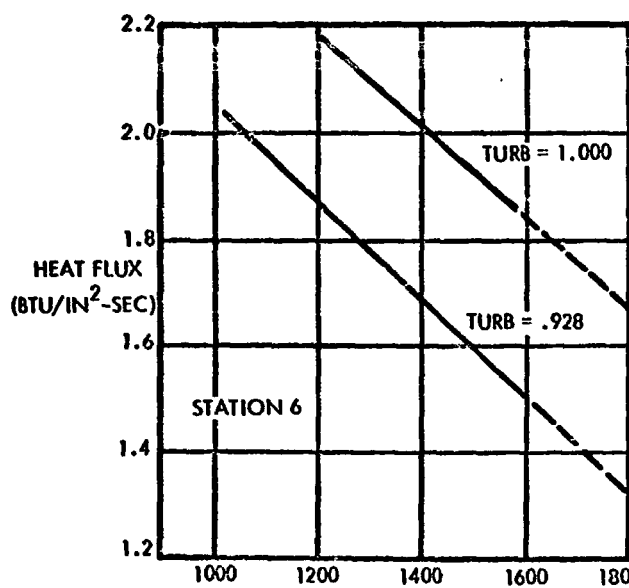


FIGURE 3-34

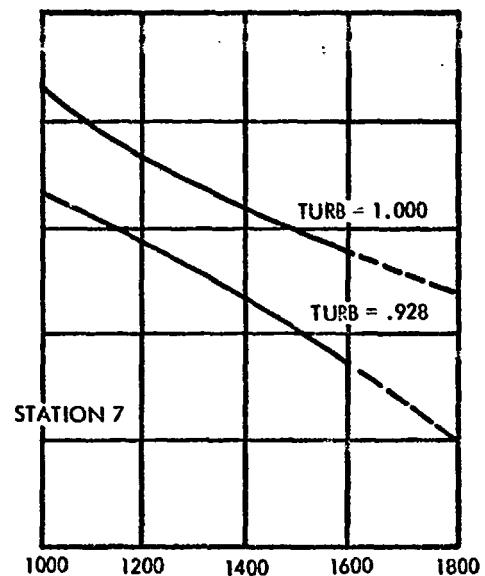


FIGURE 3-35

CONTOUR "C" HEAT FLUX LEVELS AT FTP

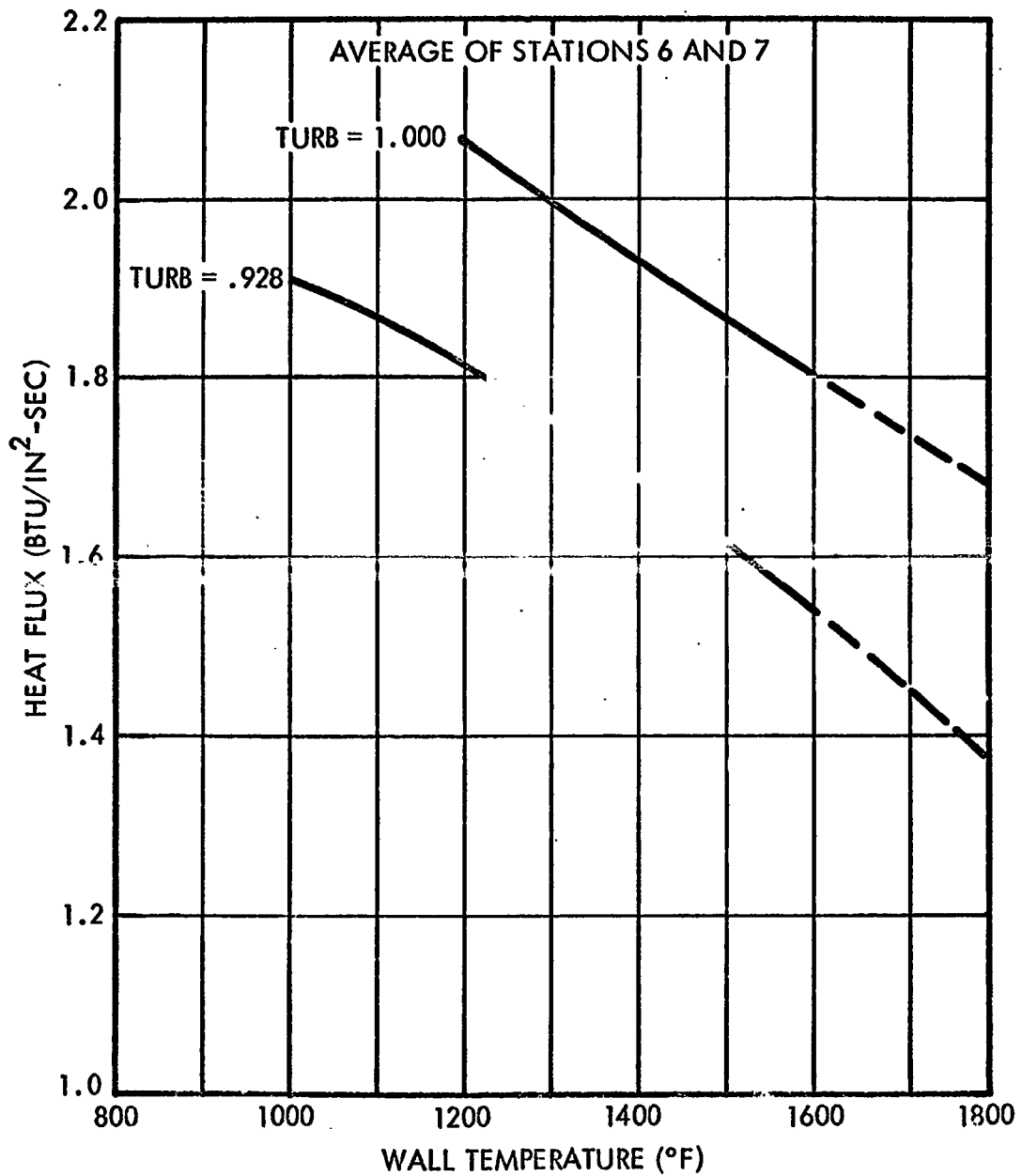


FIGURE 3-36 CONTOUR "C" HEAT FLUX LEVELS AT FTP

Thrust Level	Station	1.025 Turb.	0.928 Turb.	
		Q/A (Btu/in ² -sec)	Q/A (Btu/in ² -sec)	Percent Reduction
FTP	6	1.59	1.43	-10.1
FTP	7	1.48	1.37	- 7.4
FTP	Average of 6 and 7	1.54	1.40	- 9.1
65%	6	1.68	1.57	- 6.6
65%	7	1.40	1.25	-10.7
65%	Average of 6 and 7	1.54	1.41	- 8.4

Above Q/A values have been evaluated at a wall temperature of 1600°F.

Figures 3-37 through 3-39 summarize the above results for the region in the immediate vicinity of the throat plane; i.e., Stations 6 and 7. Figure 3-37 shows throat heat flux as a function of turbulator size for Contour "A" at FTP with two different wall temperatures. Figure 3-38 shows throat heat flux as a function of turbulator size for both of the contours tested at FTP with a wall temperature of 1600°F. Figure 3-39 presents throat heat flux as a function of turbulator size for Contour "A", a wall temperature of 1600°F, and two different thrust levels.

3.6 WATER-COOLED CHAMBERS

The following subsections describe the experimental hardware, data, analyses, and results achieved during the program for two heat transfer rigs utilizing water-cooling schemes to derive the required heat transfer data. Figure 3-40 shows the steel water-cooled chamber which has three separate complete annulus water passages. Figure 3-41 shows the aluminum water-cooled chamber. This chamber was composed of many isolated cooling slots throughout the entire chamber in order to determine heat transfer variations circumferentially as well as axially.

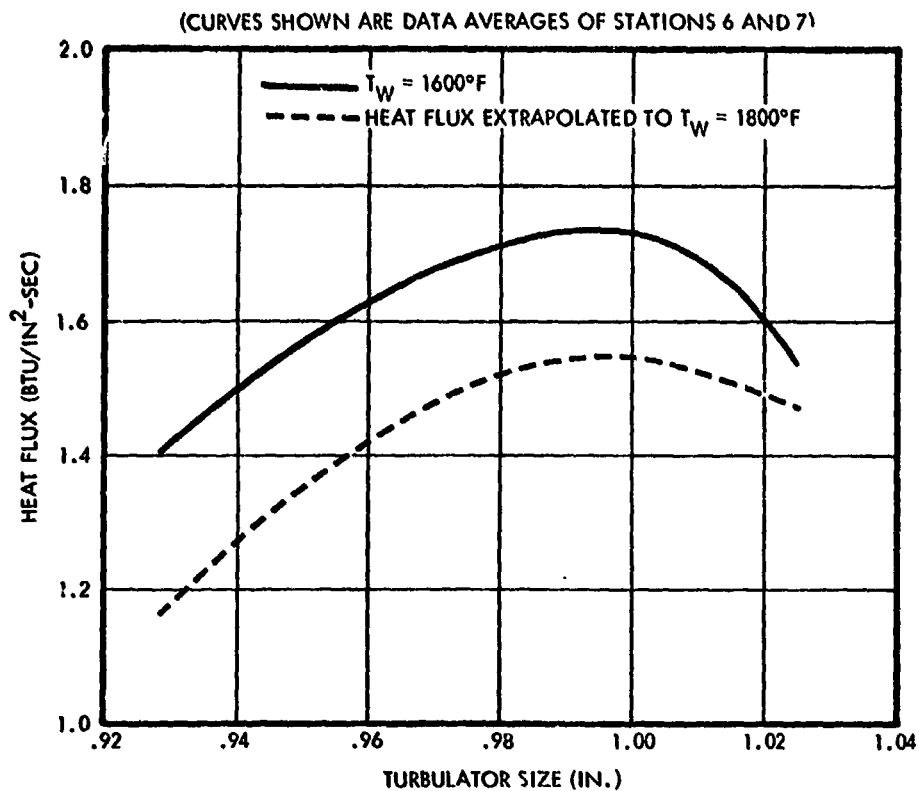


FIGURE 3-37 CONTOUR "A" HEAT FLUX AT FTP

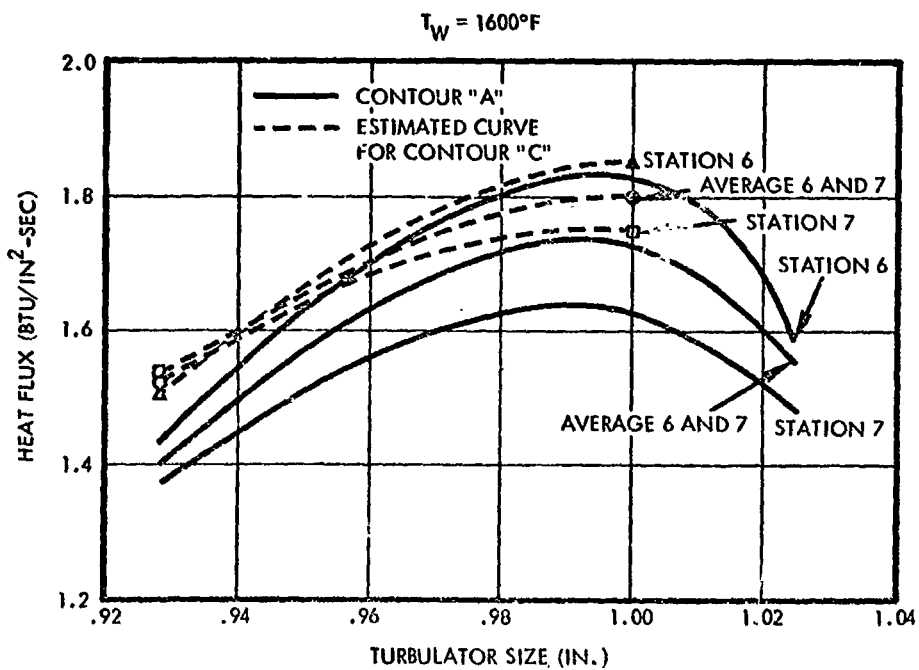


FIGURE 3-38 CONTOURS "A" AND "C" HEAT FLUX AT FTP

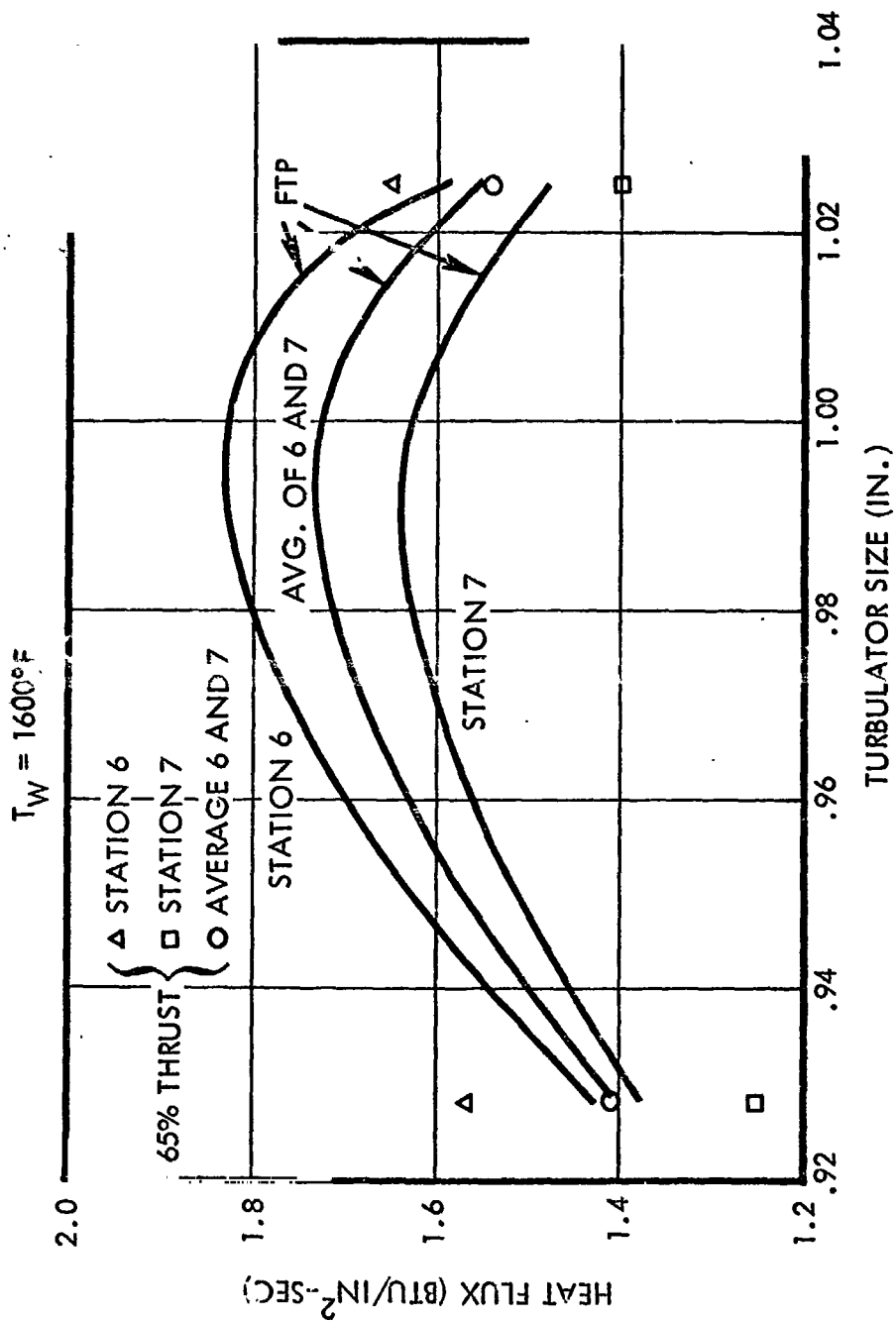


FIGURE 3-39 CONTOUR "A" HEAT FLUX AT FTP AND 65% THRUST

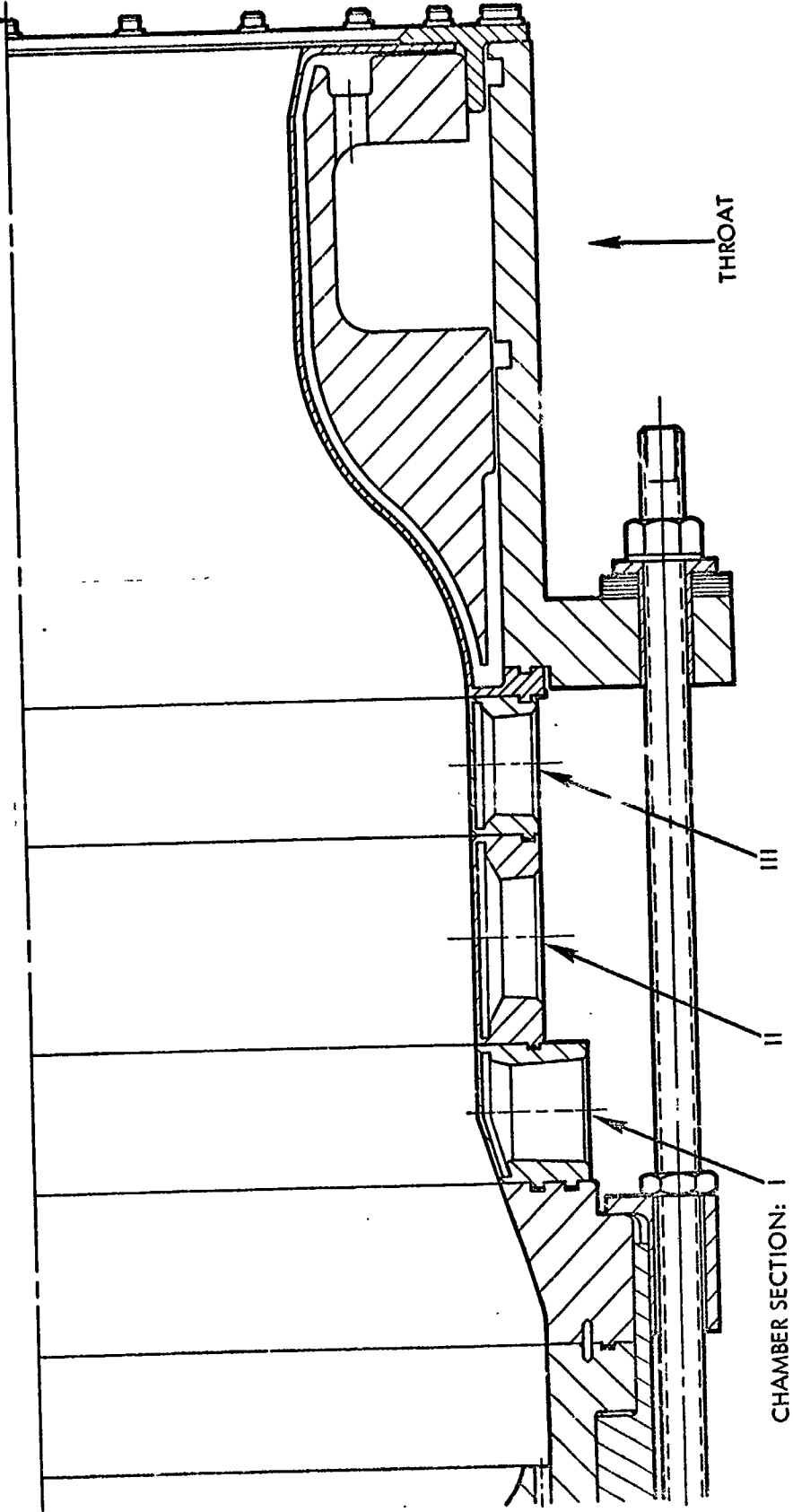


FIGURE 3-40 STEEL WATER-COOLED CHAMBER

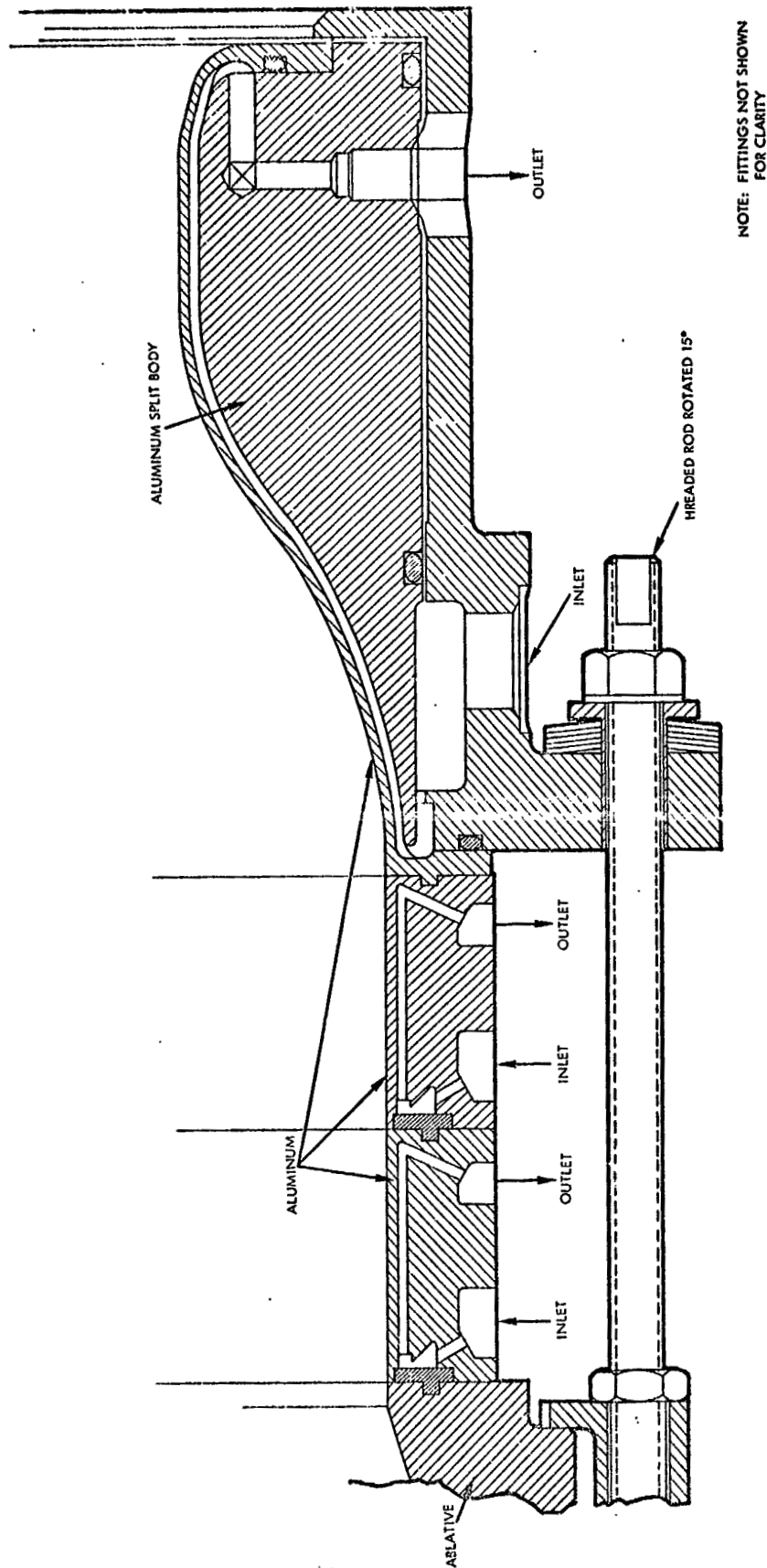


FIGURE 3-41 EXPERIMENTAL ALUMINUM WATER-COOLED CHAMBER

3.6.1 Steel Water-Cooled Chambers

A series of LMDE water-cooled steel chamber tests were performed and the results were analyzed to derive heat flux to the walls and possible correlations between recovery temperatures and film coefficients as a function of chamber axial location and thrust. These data were also used to substantiate baseline thermal design margins for the CIC Program heat transfer test chambers. The results presented represent two head end assemblies. HEA 038 was tested before HEA 023. HEA 023 was the CIC Program common head end assembly used with all chamber configurations except during the turbulator test series.

Several significant conclusions may be drawn from the analysis. These are presented below with comparisons between HEA 038 and 023 results.

- The water-cooled chamber heat flux using HEA 023 is significantly greater than when using HEA 038 at thrust levels greater than 25%.
- HEA 038 showed the throat region gas recovery temperature to be maximum around 25% thrust.
- HEA 023 tests indicate a peak recovery temperature exists in the throat region at around 25% and 80% thrust although it is not as clearly defined as for HEA 038. Furthermore throat recovery temperature could be considered as almost constant over the thrust range for HEA 023 tests.
- For the chamber barrel from the injector to approximately 5 inches downstream of the turbulence ring, the maximum heat flux to the walls occurs at a thrust level of approximately 70%. The throat maximum heat flux occurs at FTP.
- At FTP the chamber barrel experiences very low heat loads from the injector to approximately 5 inches downstream of the turbulence ring.
- Film barrier cooling effects in the barrel diminish rapidly at thrust levels below FTP.

- Due to unpredictable combustion turbulence and two-zone profiles in the chamber, conventional pipe flow correlations and Bartz prediction methods are inadequate for predicting film coefficient profiles in the chamber barrel. By using an assumed recovery temperature profile and iterating between film coefficient and recovery temperature some degree of correlation is obtained between these results and preliminary heat sink test results.
- There is some evidence from the data analysis to substantiate that the "pipeflow" correlations and Bartz's prediction methods are reasonably accurate at the throat.

3.6.1.1 Data Reduction and Analysis

The steel LMDE water-cooled chamber of Figure 3-40 is shown schematically in Figure 3-42 along with all of the test data acquired from test firings using HEA 023 and 038. The measured coolant water parameters and engine operating conditions shown were for steady-state operation.

The series of tests analyzed had the coolant flow rate and bulk temperature rise measured for the total chamber and for each of the four separate sections. Thus, average heat flux was calculated for each test as follows:

Heat:

$$Q = \dot{m}_{H_2O} \Delta T_{H_2O} \quad (5)$$

Heat Flux:

$$Q/A = \frac{Q}{\text{Area}} \quad (6)$$

HEA 023 DATA

CHAMBER COOLANT WATER PARAMETERS

HEAD END P.C. (PSIA)	PROP MR (O/F)	PIN (PSIA)	TOTAL W	W-1	W-2	W-3	CALC WTH (THROAT)	TIN (°F)	TE-1	TE-2	TE-3	TE-4	TE-5	ΔT (TOTAL)
103.0	1.60	120	67.5	7.52	5.48	5.66	48.84	68.5	79.9	96.3	87.8	81.9	80.7	14.3
70.4	1.58	120	67.4	7.47	5.42	5.72	48.79	68.5	82.5	98.1	85.5	79.2	78.4	12.6
55.4	1.60	120	67.6	7.44	5.64	5.84	48.68	68.5	79.8	91.4	82.6	77.1	76.3	10.3
45.1	1.60	120	67.6	7.38	5.77	5.88	48.57	68.5	77.7	87.2	80.5	76.2	75.4	8.8
29.0	1.59	120	67.8	7.30	6.00	5.98	48.52	68.5	73.8	78.9	76.0	74.1	73.9	6.2
14.0	1.61	120	67.8	7.21	6.15	6.14	48.30	68.5	71.9	71.7	70.9	70.0	70.1	2.2

RUN VA1-662

HEA 038 DATA

CHAMBER COOLANT WATER PARAMETERS

HEAD END P.C. (PSIA)	PROP MR (O/F)	PIN (PSIA)	TOTAL W (LB/SEC)	W-1	W-2	W-3	CALC WTH (THROAT)	TIN (°F)	TE-1	TE-2	TE-3	TE-4	TE-5	ΔT (TOTAL)
103.6	1.606	70	68.90	6.87	4.86	5.85	51.32	72.41	78.8	92.5	84.7	84.7	84.4	12.59
53.1	1.554	70	68.95	6.87	4.83	6.00	51.25	72.42	82.1	97.2	84.7	84.7	84.4	9.05
27.2	1.505	70	69.70	6.87	4.90	6.20	51.73	72.44	77.3	86.4	84.7	84.7	84.4	5.85
11.2	1.340	70	69.70	6.81	4.94	6.35	51.60	72.43	75.4	77.9	84.7	84.7	84.4	2.26
103.5	1.593	70	68.93	6.85	4.84	5.82	51.42	71.21	78.1	92.5	89.4	83.2	82.9	12.51
54.1	1.582	70	68.92	6.86	4.79	5.98	51.29	71.09	81.9	96.5	84.9	78.4	78.2	9.25
28.1	1.580	70	68.86	6.76	4.86	6.13	51.11	71.11	76.5	85.0	79.9	76.5	76.2	6.22
12.1	1.596	70	68.99	6.73	4.90	6.28	51.38	71.08	74.7	75.4	74.1	73.0	72.6	2.19

RUN VA2-529

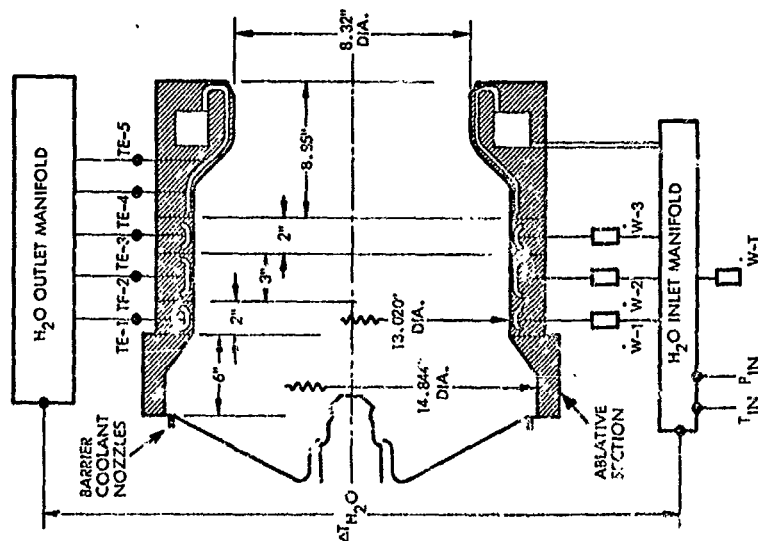
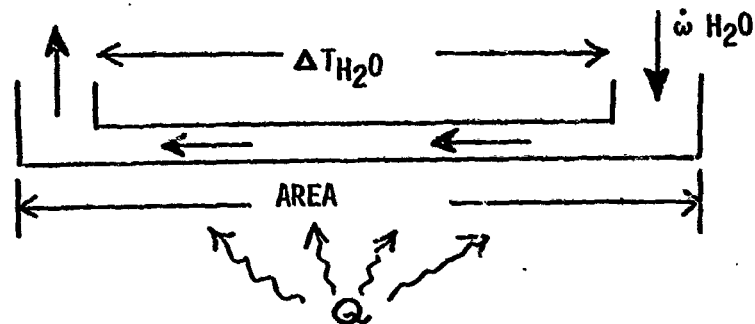
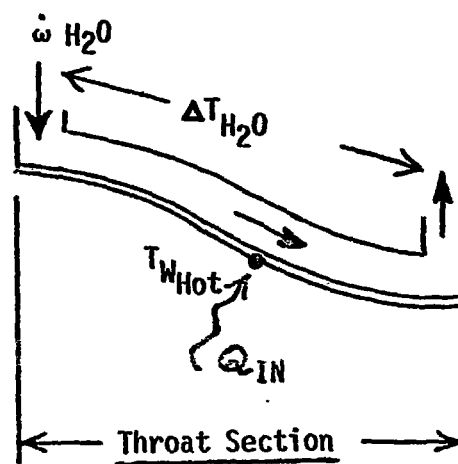


FIGURE 3-42 STEEL WATER-COOLED CHAMBER TEST DATA

Chamber Sections

In order to determine the average recovery temperatures at each section, a convective heat transfer coefficient profile was assumed to exist over the entire chamber as shown in Figure 3-46. The recovery temperature was calculated as follows:

$$T_R = \frac{0}{\int h \, dA} + T_{WHot} \quad (7)$$

Alternatively, if a recovery temperature profile is assumed, then Equation (7) can be used to derive a film coefficient profile.

The hot side average wall temperature,

$$T_{W\text{Hot}} = T_{W\text{Cold}} + \frac{Q}{A} \left(\frac{t}{k} \right) \quad (8)$$

where:

t = wall thickness

k = conductivity of wall material

The inside cold wall temperature is obtained by assuming that heat transfer to the water flowing in the wall cooling annulus is taking place under conditions of sub-cooled nucleate boiling. If T_{SAT} is the saturation temperature of the liquid, then under these conditions the maximum value of $(T_{W\text{Cold}} - T_{\text{SAT}})$ is 60°F and the minimum is about 30°F. For purposes of this analysis it was assumed that $(T_{W\text{Cold}} - T_{\text{SAT}})$ was 50°F under all conditions. The saturation temperature of the water under test conditions was 300°F.

Therefore,

$$T_{W\text{Cold}} = 350^\circ\text{F}$$

Now,

$$t/k = 260$$

$$\left\{ \begin{array}{l} \text{Wall material: 347 SS} \\ \text{Wall thickness: 0.065 in. (all sections)} \end{array} \right.$$

Hence,

$$T_{W\text{HOT}} = 260 \frac{Q}{A} + 350 \quad (9)$$

3.6.1.2 Thermal Environment Results and Conclusions

The measured steady-state test data from Figure 3-42 was reduced to thermal environment information and the significant results are as follows:

- Figures 3-43, 3-44, and 3-45 show the calculated average heat flux at each water-cooled section as a function of chamber pressure.
- Figure 3-46 shows the Bartz theoretical film coefficients (from Section 3.4) for the LMDE chamber and throat as a function of chamber pressure.

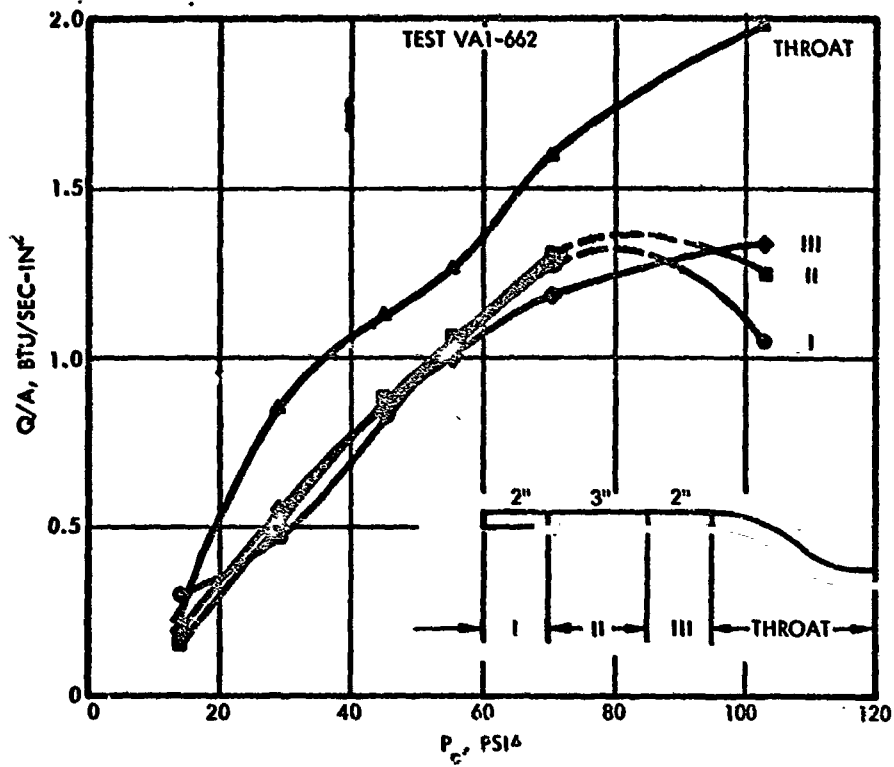


FIGURE 3-43 AVERAGE HEAT FLUX (HEA 023)

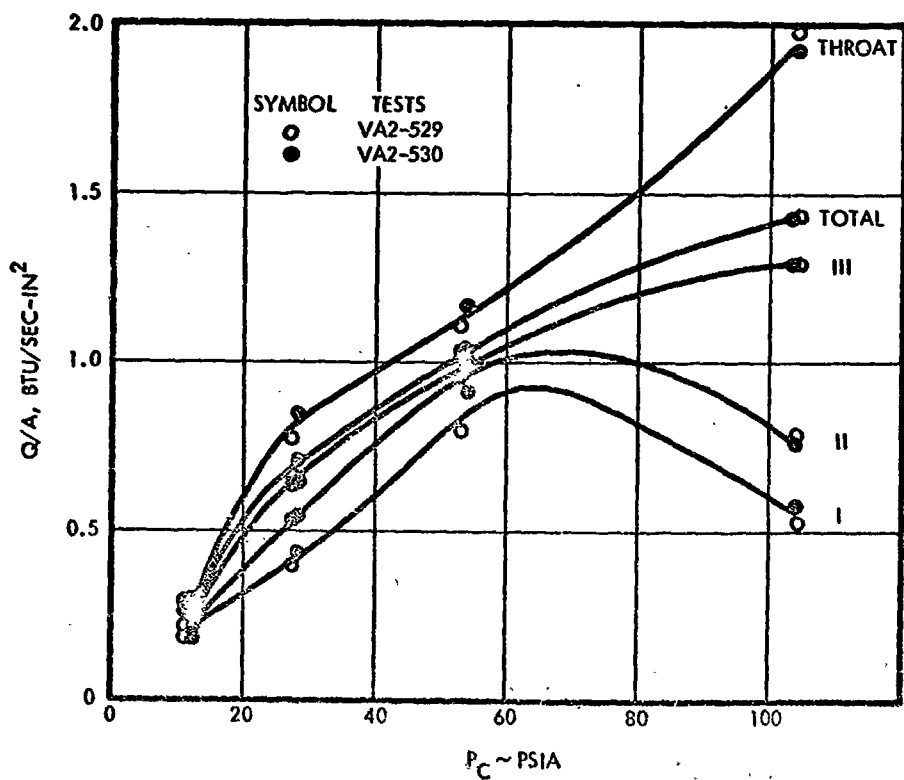


FIGURE 3-44 AVERAGE HEAT FLUX (HEA 038)

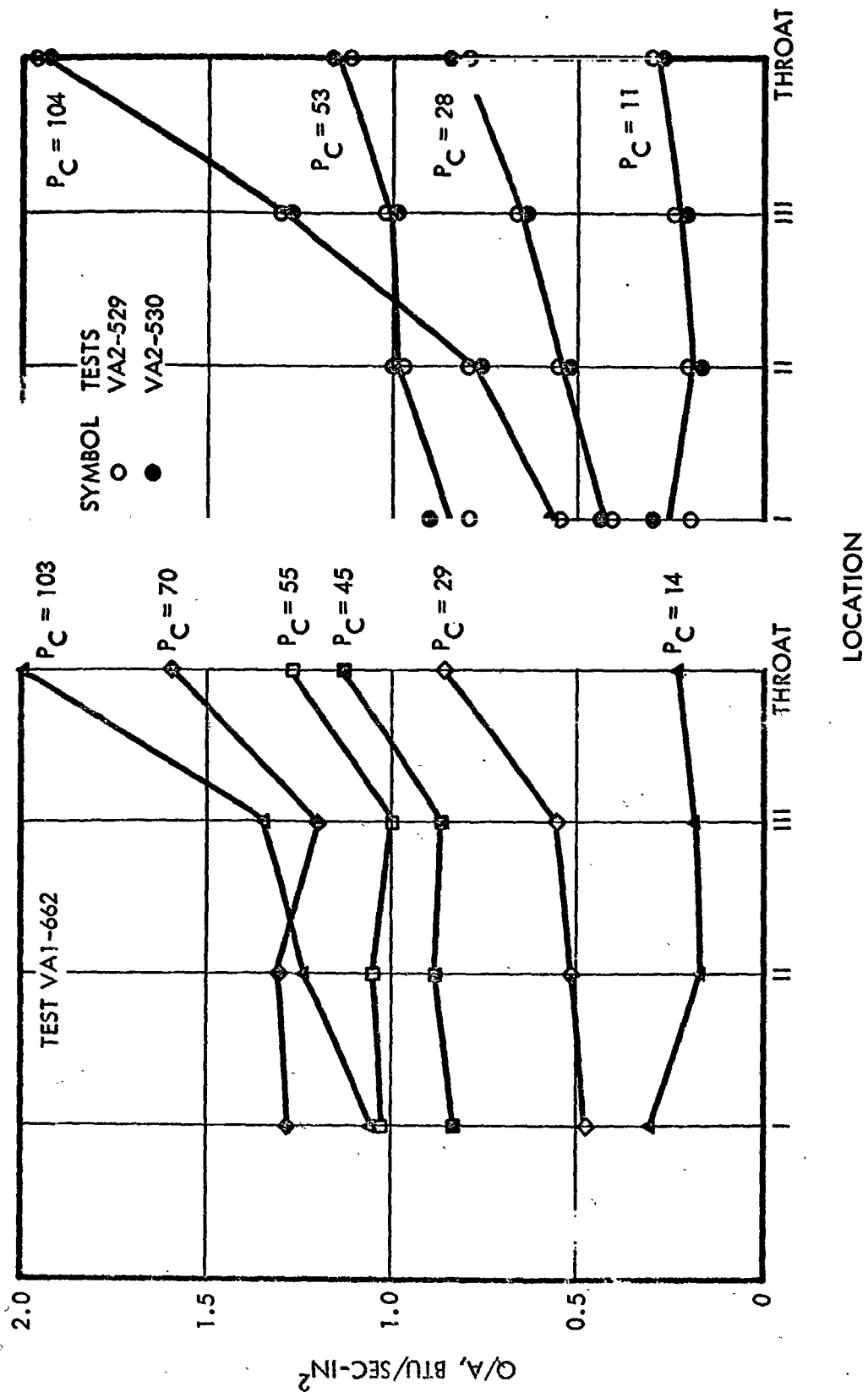


FIGURE 3-45 AVERAGE HEAT FLUX (HEA 023 AND 038)

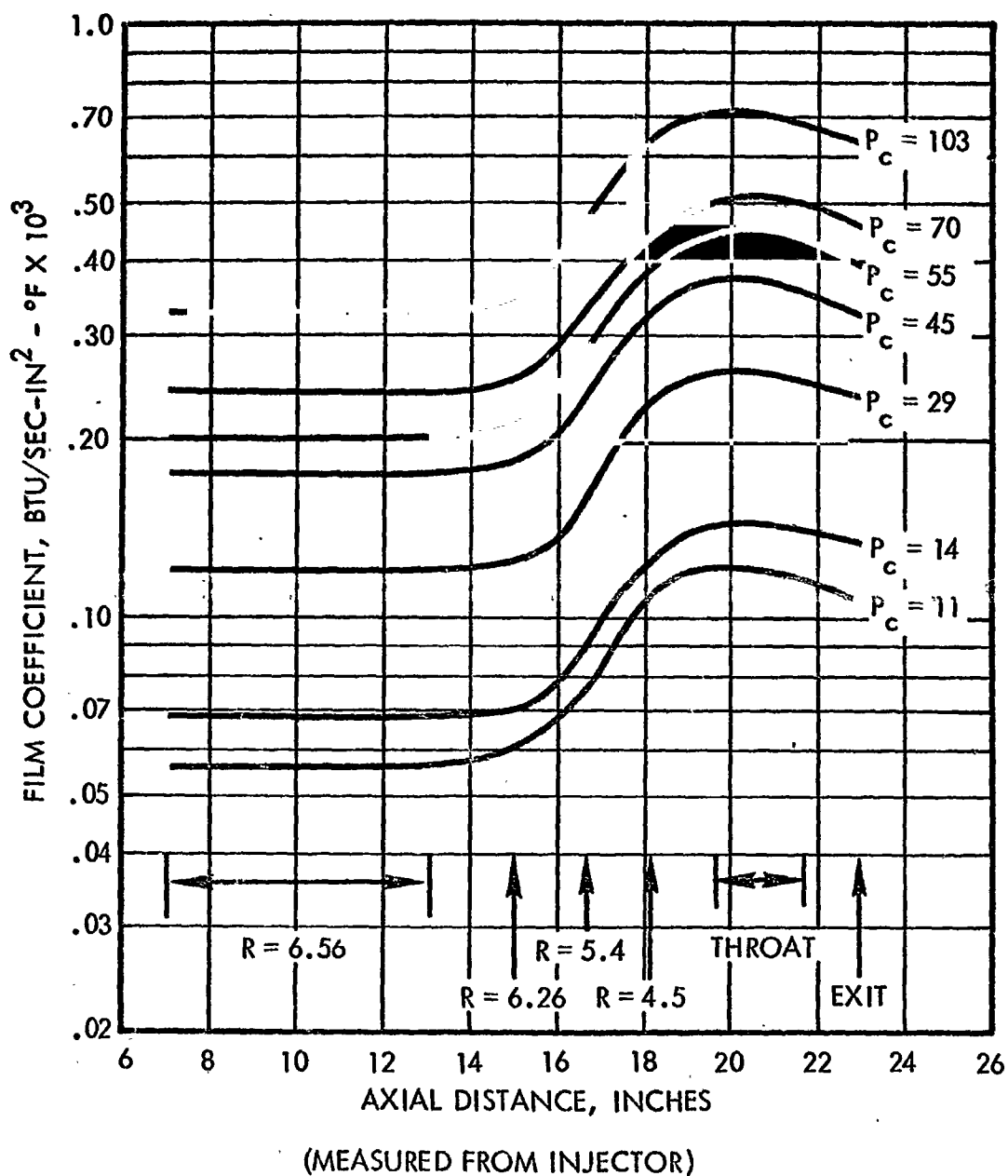


FIGURE 3-46 BARTZ THEORETICAL FILM COEFFICIENTS

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- Table 3-5 shows the estimated hot side wall temperatures for each chamber section as a function of chamber pressure using Equation (9).

TABLE 3-5. ESTIMATED HOT SIDE WALL TEMPERATURES

HEA 023, Test VA2-662 Data			HEA 038, Tests VA-529 and 530 Data		
Chamber Pressure (psia)	Location	T _{W Hot} (°F)	Chamber Pressure (psia)	Location	T _{W Hot} (°F)
103	I	623	104	I	493
	II	672		II	550
	III	697		III	688
	Throat	870		Throat	857
70	I	683	53	I	571
	II	690		II	602
	III	660		III	607
	Throat	765		Throat	650
55	I	618	28	I	460
	II	623		II	490
	III	613		III	520
	Throat	680		Throat	565
45	I	565	11	I	410
	II	580		II	410
	III	575		III	410
	Throat	645		Throat	420
29	I	428			
	II	480			
	III	490			
	Throat	570			
14	I	428			
	II	392			
	III	397			
	Throat	410			

- Table 3-6 shows the average coolant flow parameters and heat transfer areas for the test rigs.
- Figures 3-47 and 3-48 show the calculated recovery temperatures for each chamber section as a function of chamber pressure using nominal values of convective film coefficients from Bartz's short form analysis.
- Figures 3-49 and 3-50 show the calculated recovery temperatures for each section as a function of chamber pressure. Two sets of recovery temperature curves are shown. The higher set represents the results from using the nominal Bartz coefficients ($BARTZ = 1$). The lower set shows the reduction of calculated recovery temperature for Bartz coefficients that are twice the nominal value ($BARTZ = 2$). The results show that the conventional forms of the film coefficient profile do not predict expected recovery temperature profiles axially in the chamber. Since a recovery temperature greater than 5150°F (combustion temperature) is not possible in the LMDE chamber, the assumption of nominal Bartz coefficient levels for the chamber is erroneous for the full chamber pressure range investigated.
- Figures 3-51, 3-52, and 3-53 present an analysis of the data for HEA 023 tests in a slightly different form. The recovery temperature is assumed constant regardless of chamber location or pressure and the heat transfer coefficient is evaluated as a function of location and chamber pressure. The figures present the results for recovery temperatures of 5000°F , 4000°F and 3000°F and the heat transfer coefficient is presented as a function of the nominal Bartz coefficient at each location. The coefficient ratios shown represent the amplification factor relative to Bartz.
- Neither the assumption of $BARTZ = 1$ or a constant recovery temperature is strictly valid. Both h/h_{BARTZ} and recovery temperature will vary as a function of both chamber location and chamber pressure. Figures 3-54 and 3-55 present the results of an analysis of this effect and show the variation of recovery temperature with chamber location and pressure for an assumed film coefficient profile.

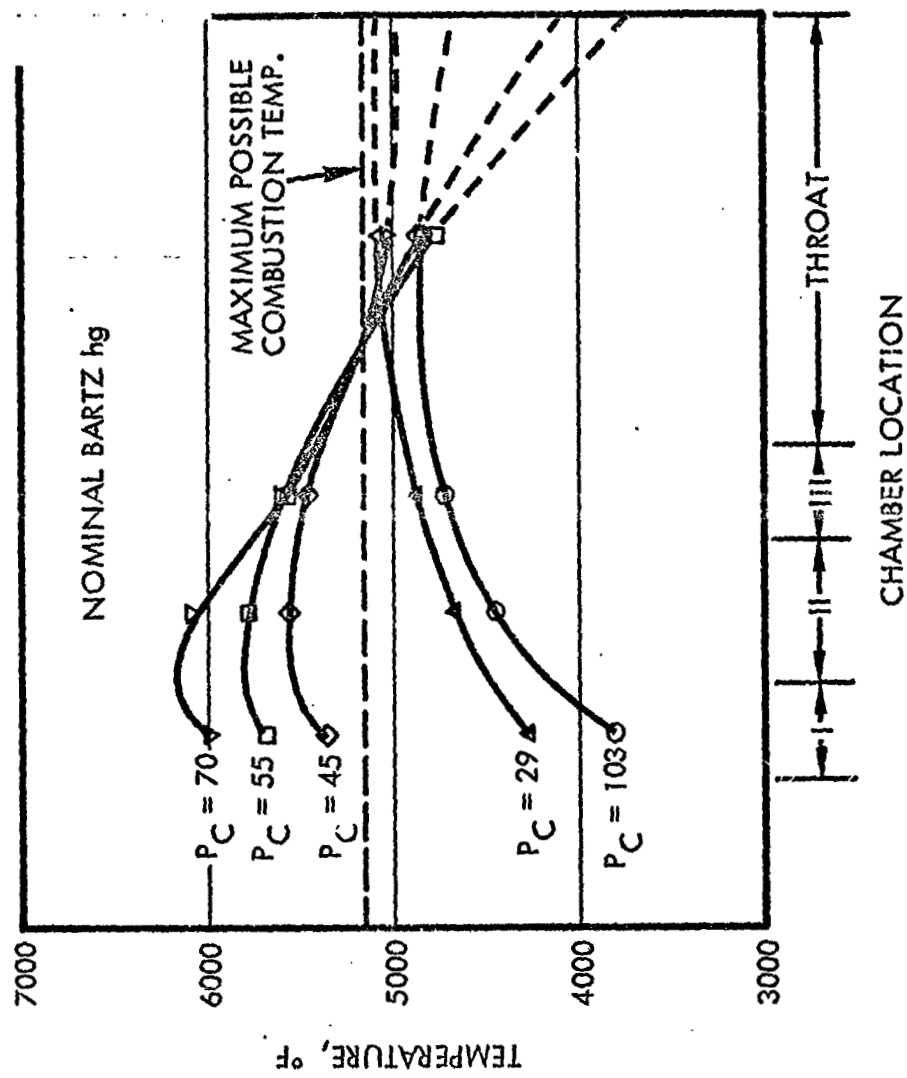


FIGURE 3-47 RECOVERY TEMPERATURE PROFILE (HEA 023)

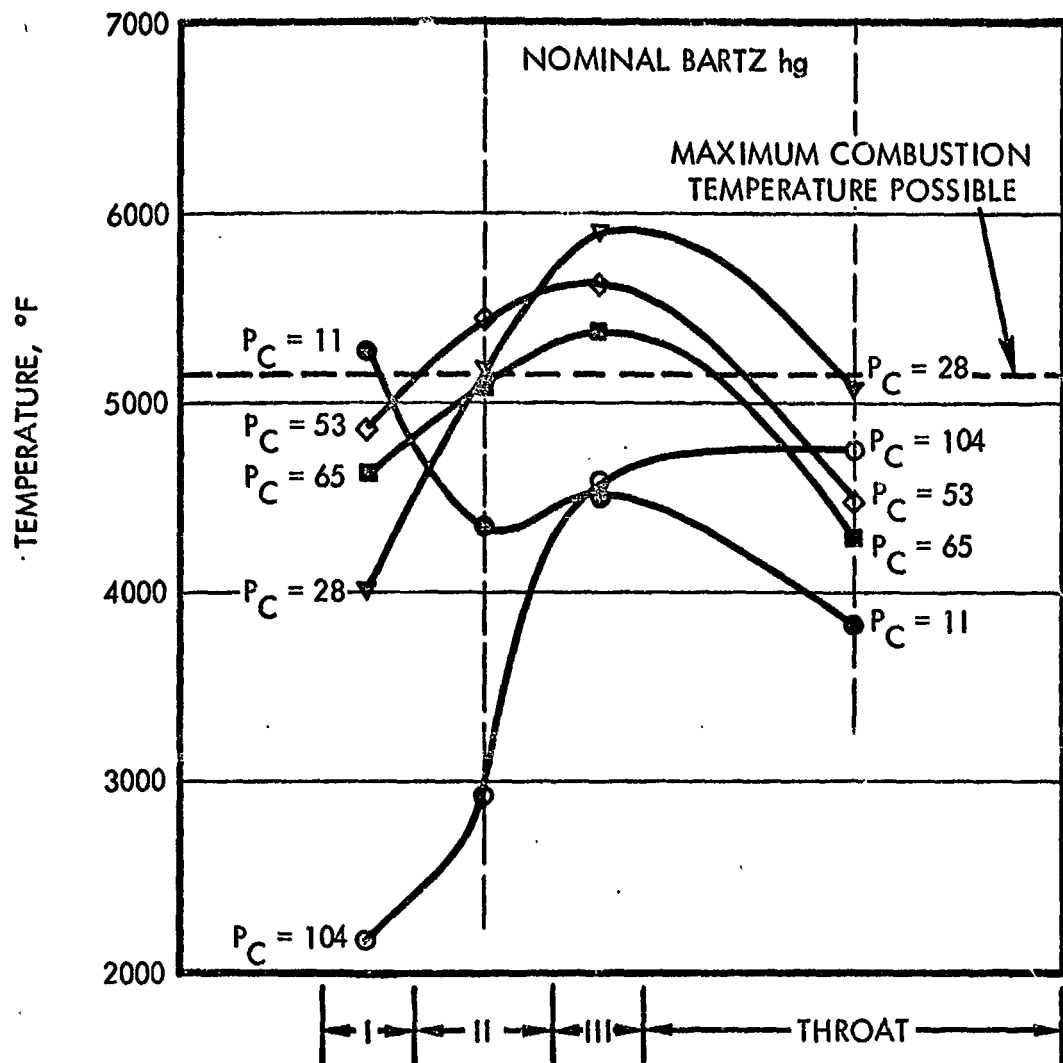


FIGURE 3-48 RECOVERY TEMPERATURE PROFILE (HEA 038)

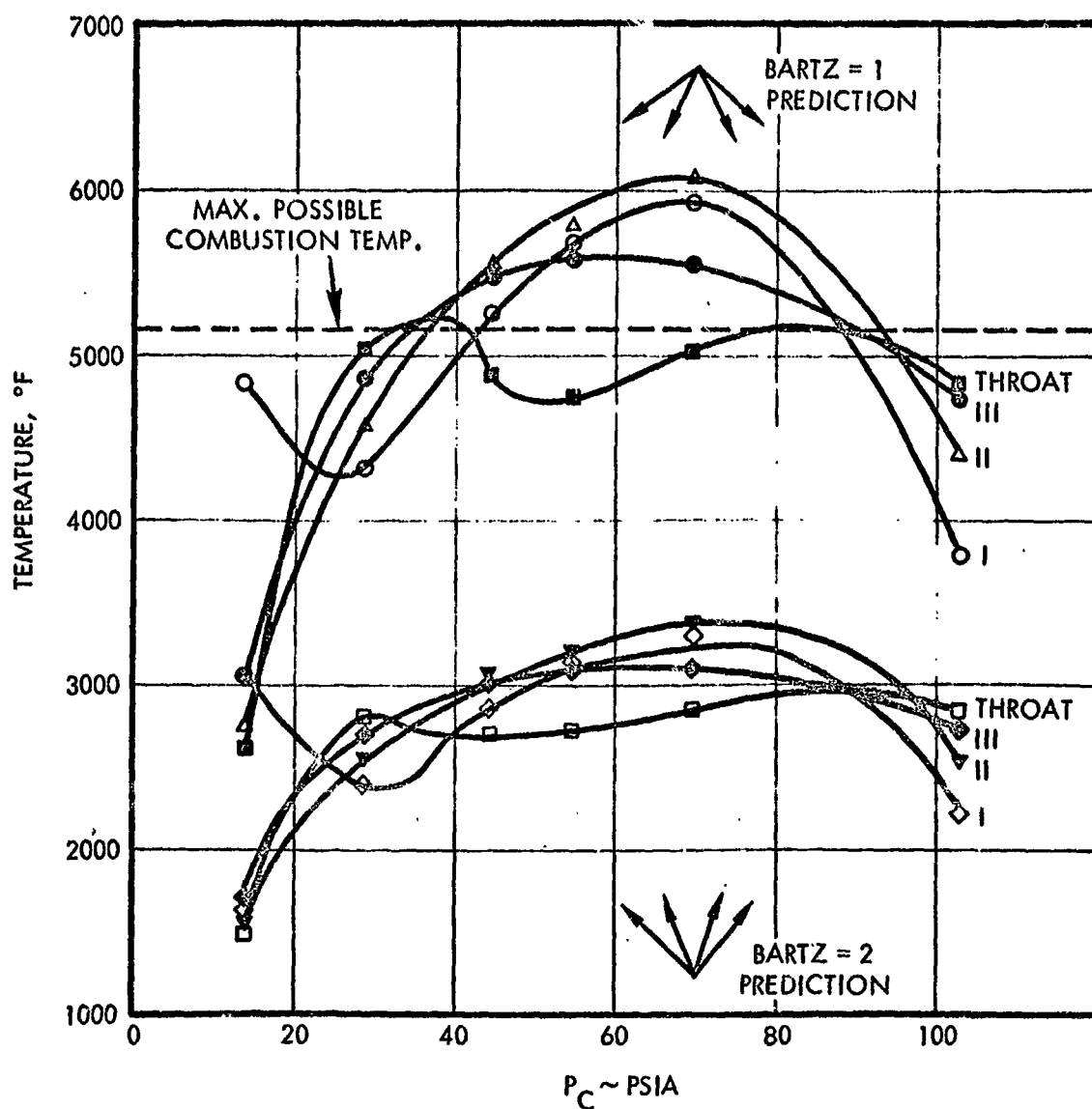


FIGURE 3-49 RECOVERY TEMPERATURE PROFILE (HEA 023)

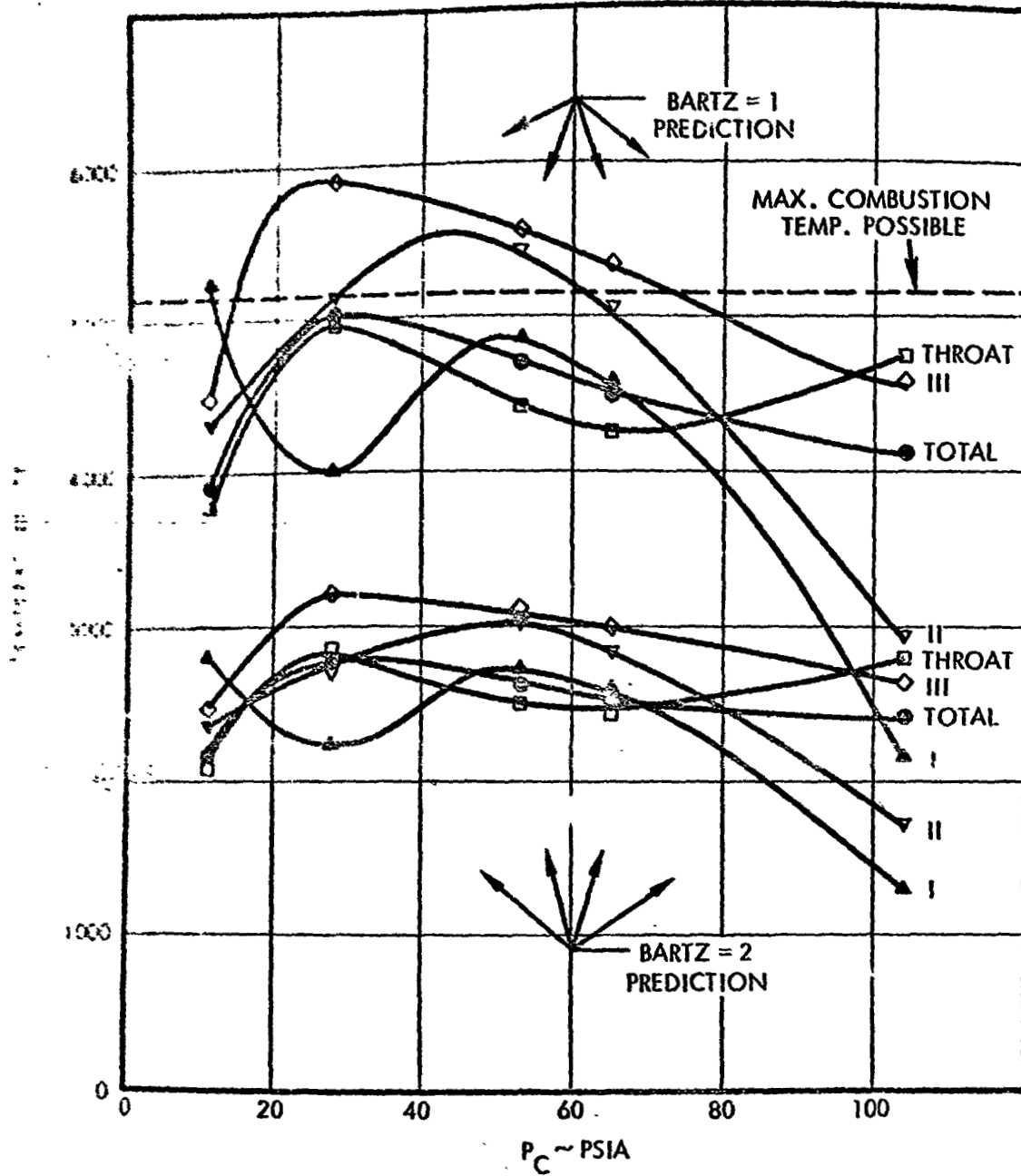
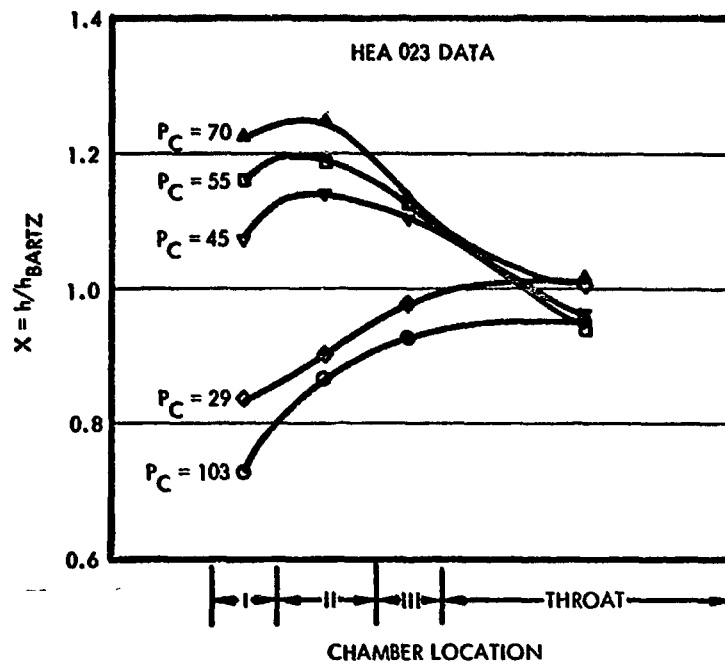
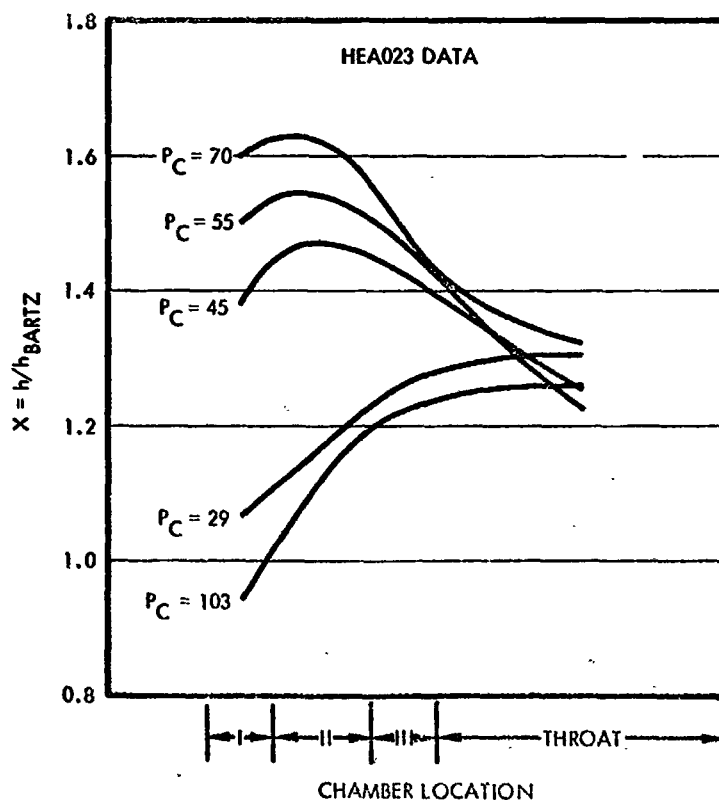
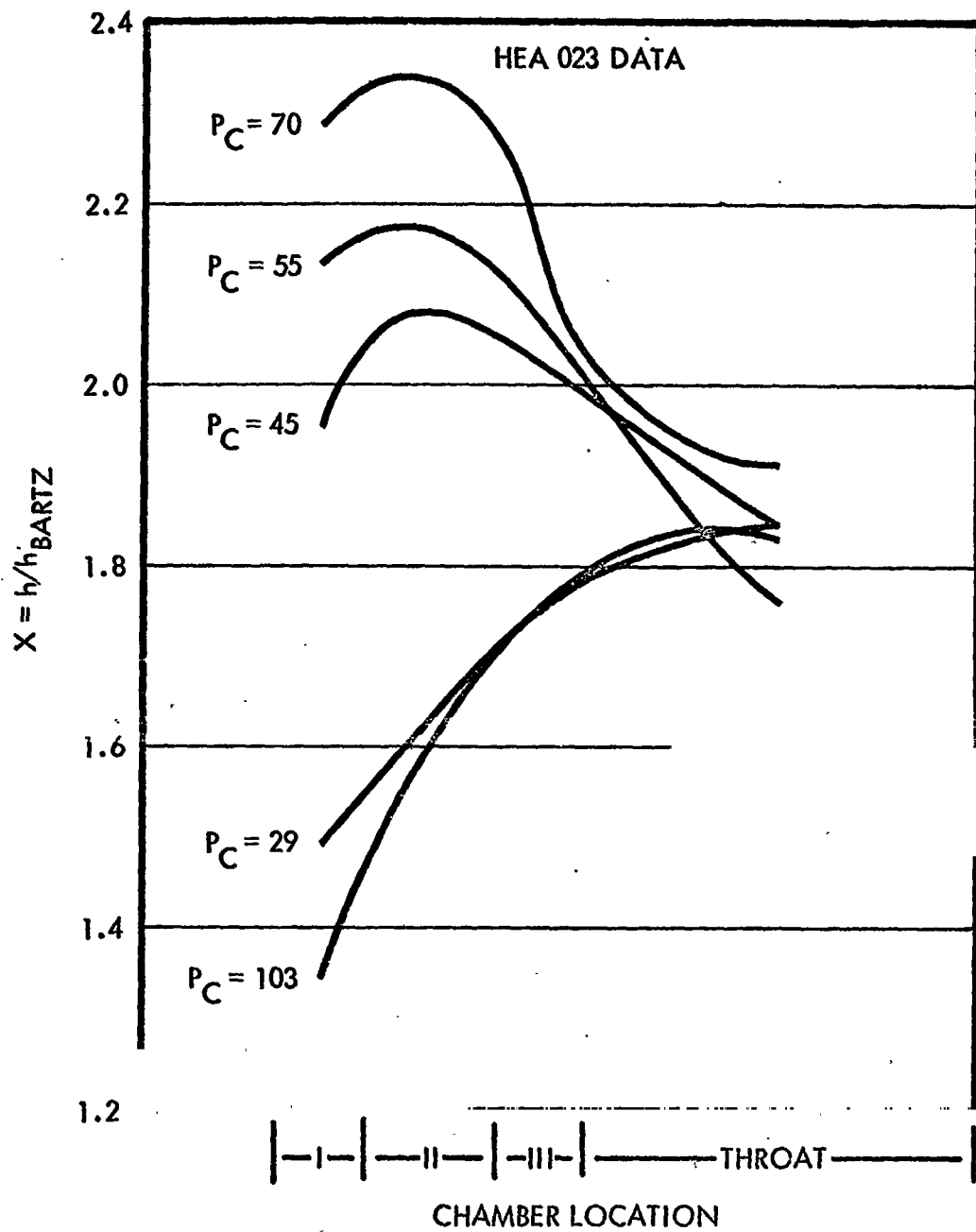


FIGURE 3-50 RECOVERY TEMPERATURE PROFILE (HEA 038)

FIGURE 3-51 AMPLIFICATION FACTORS PROFILE, $T_R=5000^\circ\text{F}$ FIGURE 3-52 AMPLIFICATION FACTOR PROFILE, $T_R=4000^\circ\text{F}$

FIGURE 3-53 AMPLIFICATION FACTOR PROFILE, $T_R=3000^\circ\text{F}$

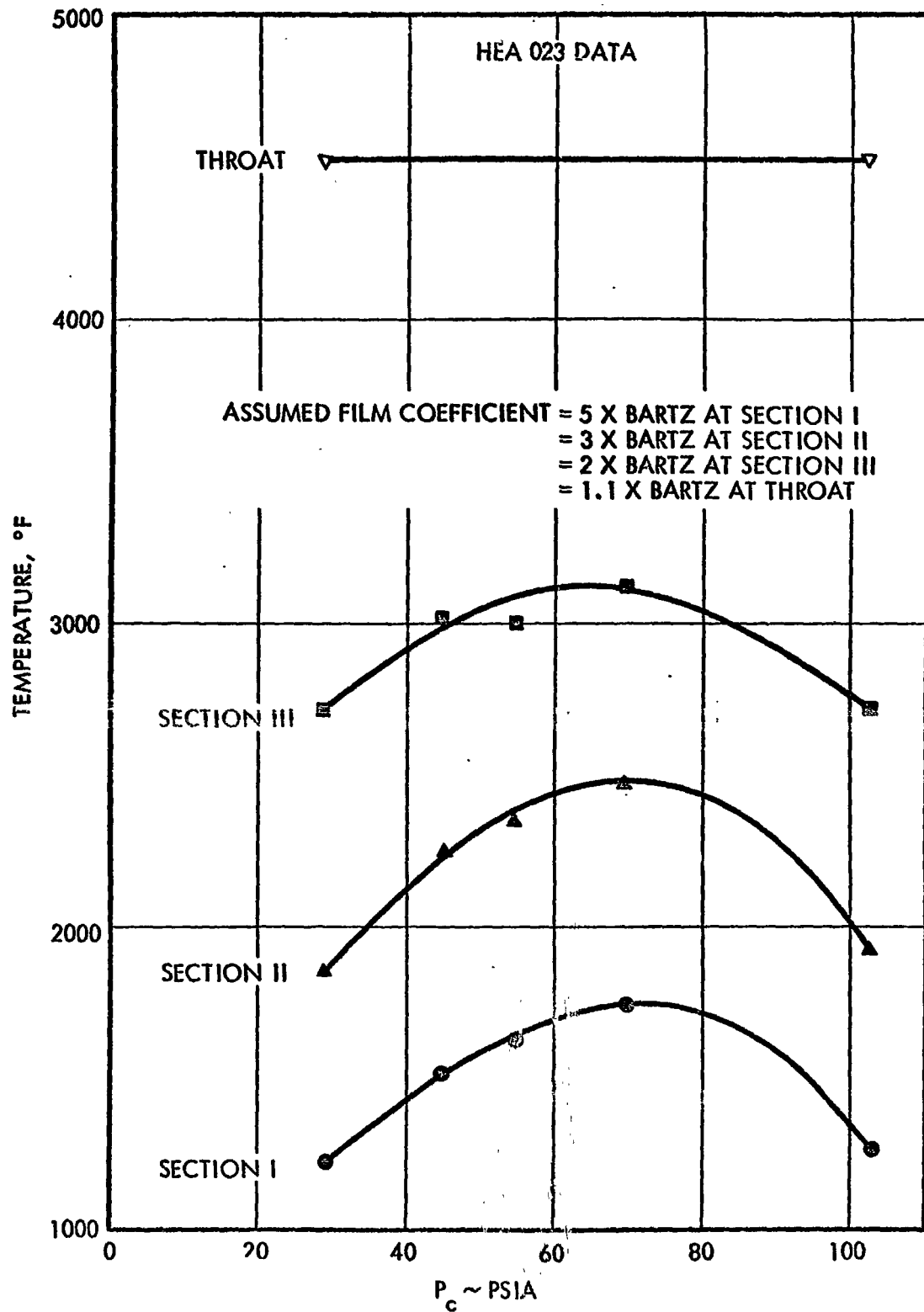


FIGURE 3-54 RECOVERY TEMPERATURE PROFILES

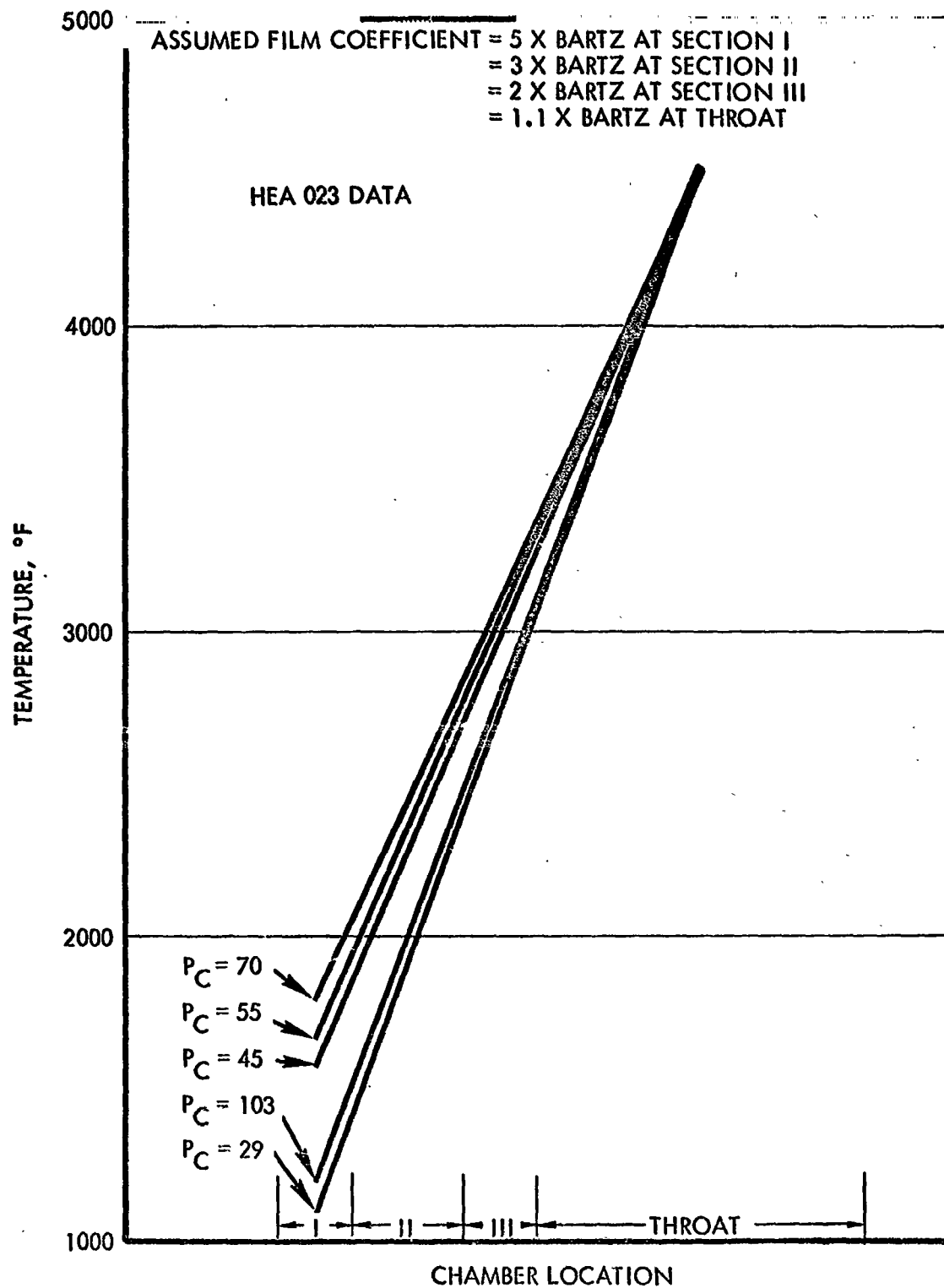
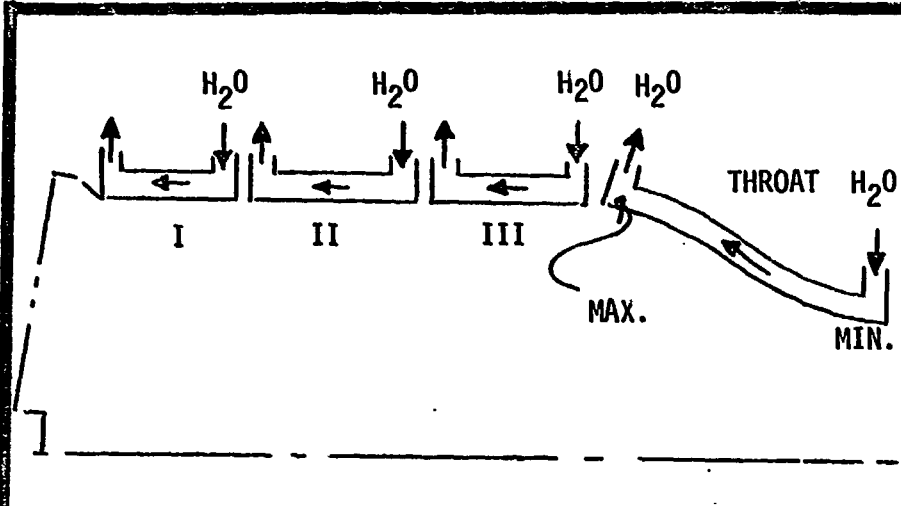


FIGURE 3-55 RECOVERY TEMPERATURE PROFILES

This assumed film coefficient is based upon iterating between a preliminary assumed temperature profile and the resultant film coefficient profile using some estimates for recovery temperatures in chamber section No. 1 based upon data from heat sink tests.

TABLE 3-7. COOLANT FLOW PARAMETERS



Section	* A_{Flow} (in ²)	** $\dot{w}_{H_2O}$ (lb/sec)	V_{H_2O} (ft/sec)
I	0.462	6.80	35.0
II	0.712	4.85	16.0
III	0.462	6.0	30.0
Throat at Minimum	6.05	51.5	20.0
Throat at Maximum	9.38	51.5	12.5

*Cooling flow-area data derived from the following TRW Systems reference prints:

E110224

E110225 - Throat Section Linear

E110226

E401253 - Assembly

EF112279 - Chamber Section Liner

** $\dot{w}_{H_2O}$ represents the average flow rate at each section for both tests.

3.6.2 Qual B Experimental Aluminum Chamber Thermal Analysis

A comprehensive thermal analysis was performed on the experimental water-cooled LMDE thrust chamber of Figure 3-56. This study was made to provide adequate thermal design margin for the aluminum heat transfer rig and to establish the water flow requirements for accurate heat transfer measurements over the full engine operating range. A hot wall temperature limit of 550°F was established in order to ensure overall chamber integrity and durability. Water flows were selected for each cooled section in order to provide a baseline for the detailed design of the water cooled chamber rig. Each of the two (2) chamber cylindrical sections were found to be safe utilizing water flow rates of 7 lb_m/sec. The throat section was adequate using 20 lb_m/sec.

Heat Transfer Data Acquisition Requirements

The heat transfer test rig was designed to provide heat transfer data to local areas of the chamber walls during steady-state engine operation. In order to acquire accurate heat transfer information the water flow rate must be sufficiently low to allow reasonable water temperature increases to occur while passing through each local area. The temperature rise must be of sufficient magnitude at the full thrust operating point in order not to compromise accuracy at low thrust points due to the lower heat input along the wall. Figures 3-57 and 3-58 show the calculated water temperature rise for the chamber and throat sections for the assumed range of conditions. It is apparent that the lower water flows yield the most temperature rise for any operating environment. It was desirable, however, that the water temperature at each section outlet not be near the boiling point at atmospheric pressure. The restriction of non-boiling at the outlet orifice prevents orifice flow characteristic changes since the water density and Reynold's number remain essentially unchanged during the tests.

Cooling Capacity Requirements

The water heat flux capability requirements were determined for each section of the chamber such that high wall temperatures would not exist in the water cooled rig during hot firing tests. Previously, LMDE test data

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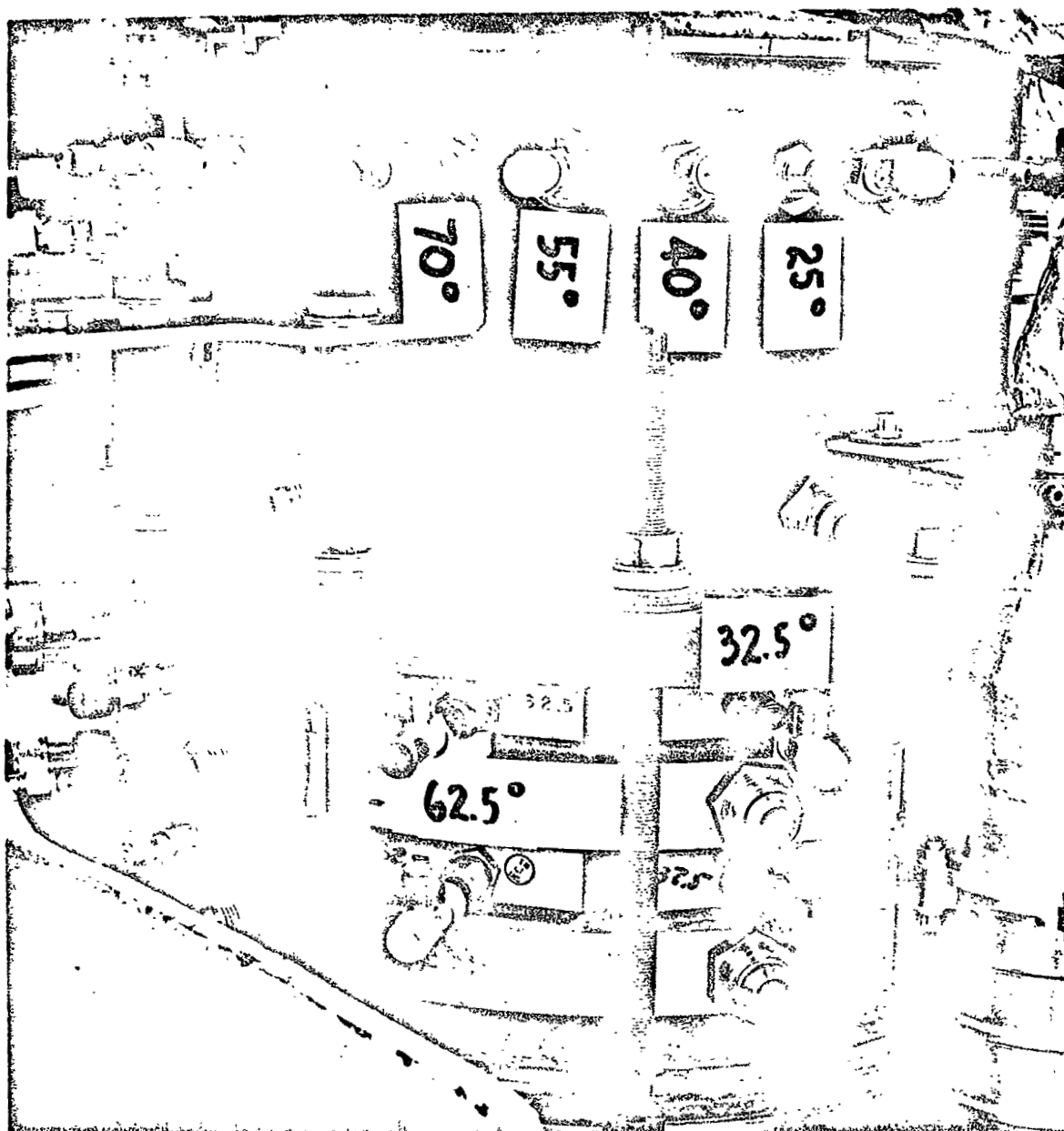


FIGURE 3-56 ALUMINUM WATER COOLED TEST RIG PHOTOGRAPH

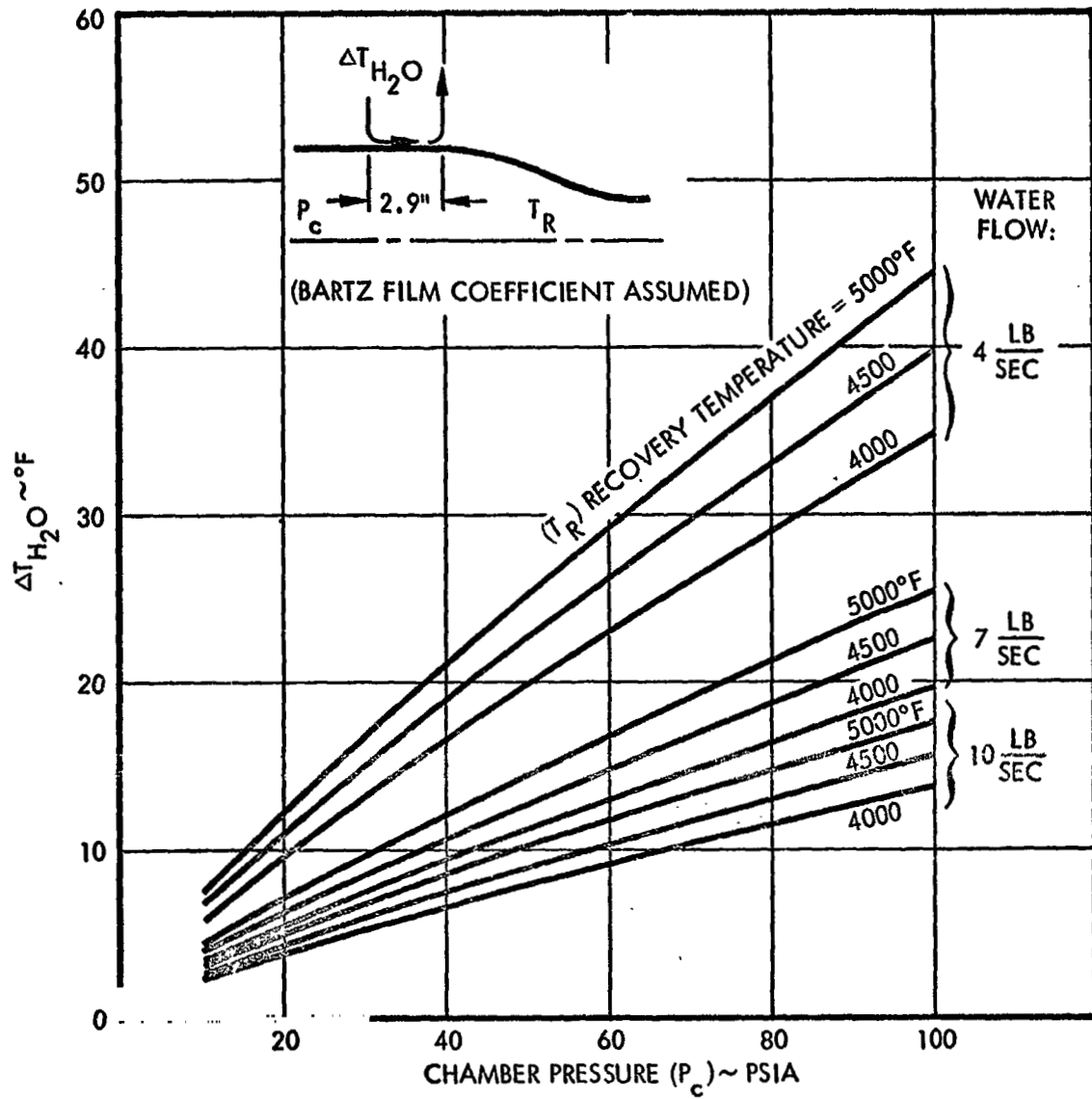


FIGURE 3-57 CALCULATED WATER TEMPERATURE RISE FOR CHAMBER SECTION

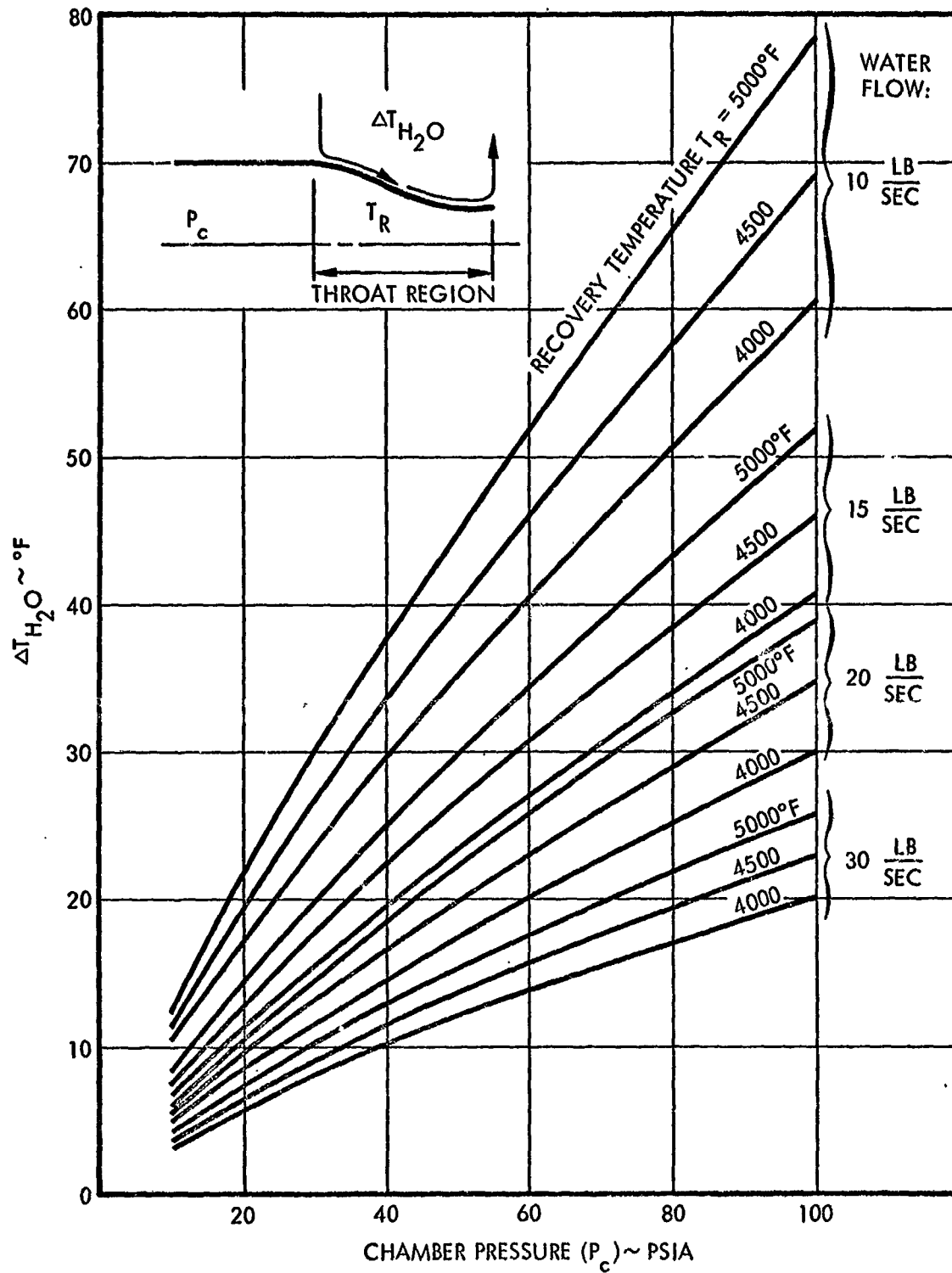


FIGURE 3-58 CALCULATED WATER TEMPERATURE RISE FOR THROAT SECTION

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showed that heat fluxes up to 3.2 Btu/sec-in^2 could be experienced locally in the chamber-throat region. For water heat flux capabilities in this region a number of forced-convection non-boiling and nucleate boiling investigations were performed. A correlation of the pertinent nucleate boiling data is shown in Figure 3-59 representing the pertinent range necessary for analyses of the water-cooled rig under maximum heat flux conditions.

Utilization of the improved cooling capabilities of water in forced-convection nucleate boiling permits the use of lower flow rates in the various sections of the test rig without compromising the maximum operating wall temperatures. Furthermore, the lower flows will promote greater heat transfer measurement accuracy due to the resulting higher temperature increase while passing through each cooled section of the rig.

The present water source supplies water at 70 to 80 psia to the rig manifold supply ring. The mean water pressure in the rig coolant passages was estimated to be 60 psia. Under this condition the water saturation temperature is 300°F . Since the water temperature varies from 70°F to over 100°F through the coolant passages, the degree of subcooling varies from 230°F to 200°F .

The anticipated coolant water velocities were derived from Figure 3-61 as follows:

Location	Flow Rate (lb _m /sec)	Velocity (ft/sec)
Chamber Cylindrical Section	7.0	11.0
Throat:		
Inlet to Transition Point	20.0	16.7
Transition Point to Exit	20.0	25.8

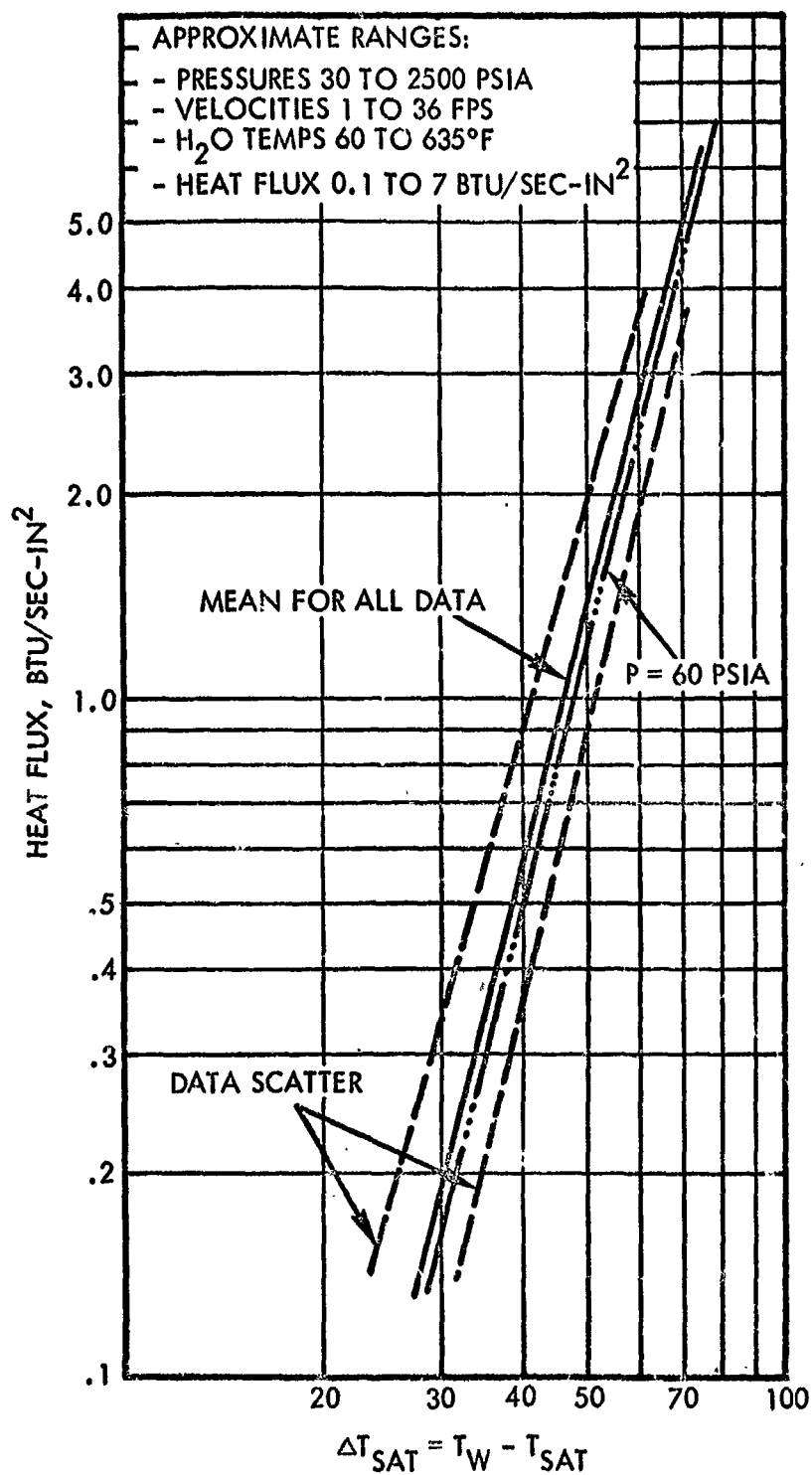


FIGURE 3-59 NUCLEATE BOILING DATA FOR WATER

The anticipated heat flux values were as follows:

Location	Anticipated Heat Flux Btu/sec-in ²	
	* Maximum	** Minimum
Each Chamber Cylindrical Section	1.35	0.15
Inlet to Throat Section	1.35	0.15
Transition Point	1.95	0.21
Throat Point	3.20	0.33

* Maximum based on $T_R = 5000^\circ\text{F}$ $P_C = 100$ psia $O/F = 1.6$

** Minimum based on $T_R = 3500^\circ\text{F}$ $P_C = 10$ psia $O/F = 1.6$

Previous heat transfer data from the LMDE steel water-cooled chamber assembly indicated that the above anticipated maximum and minimum heat flux values should adequately represent the range of conditions necessary for thermal design of the proposed water-cooled test rig. Although LMDE water test data substantiates the present thermal design criteria, additional effort was continued to isolate and define the unconventional heat transfer shifts in the LMDE chamber during duty cycle excursions or at thrust levels lower than FTP. It was not anticipated that these shifts would lower the present built-in thermal design margin for the aluminum water-cooled test rig.

Standard heat flux correlations from which Figures 3-59 and 3-60 were derived show that water nucleate boiling would exist for the maximum anticipated heat loads and water supply scheduled for the test rig. Figure 3-60 shows the peak heat flux capability as a function of coolant velocity. It is apparent that a very drastic reduction in flow would be required in order to exceed the peak heat flux and cause a burnout failure. The burnout water velocity is less than 1.5 ft/sec for the test rig at the maximum anticipated throat heat load.

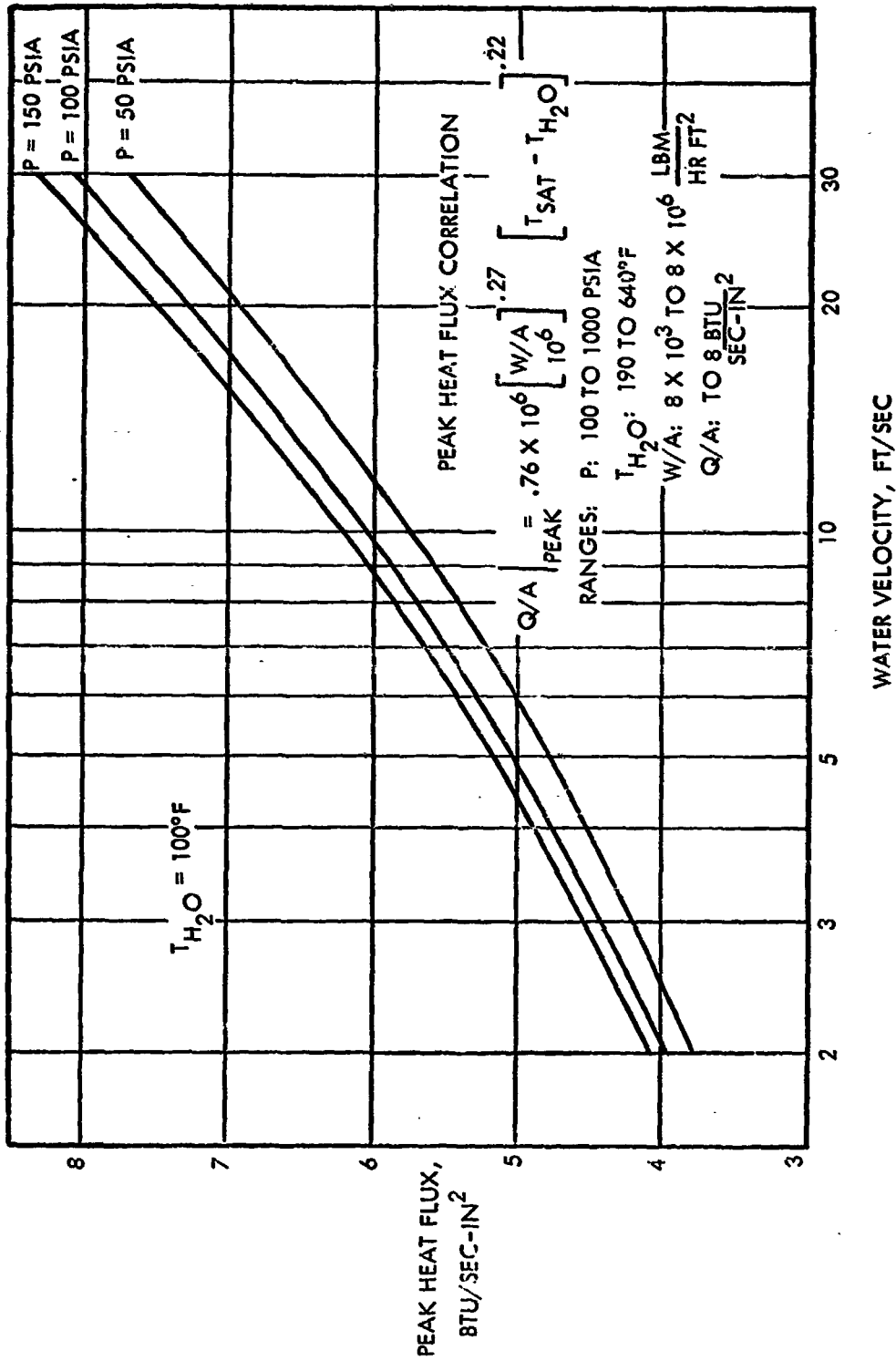


FIGURE 3-60 WATER PEAK HEAT FLUX CAPABILITY

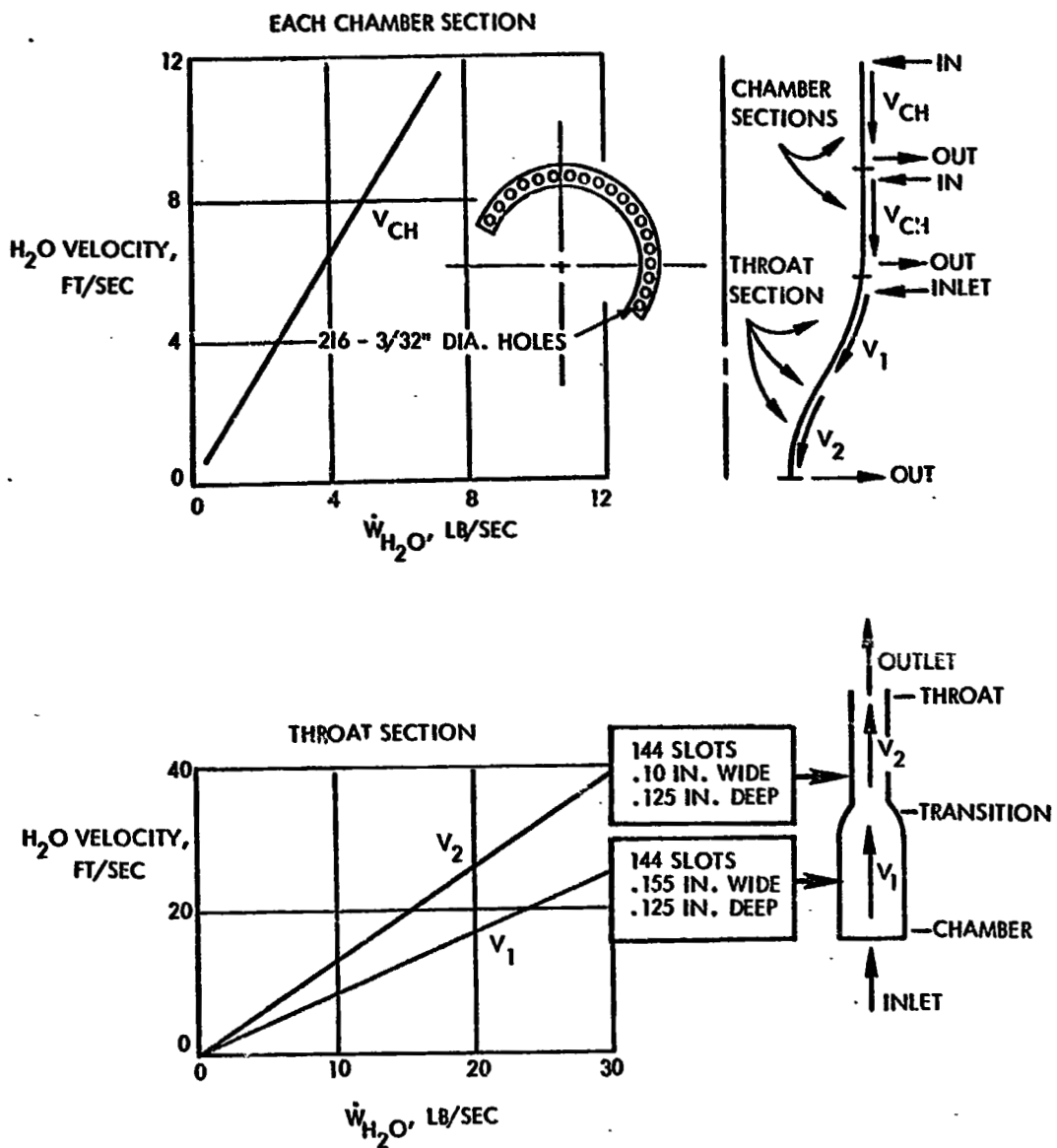


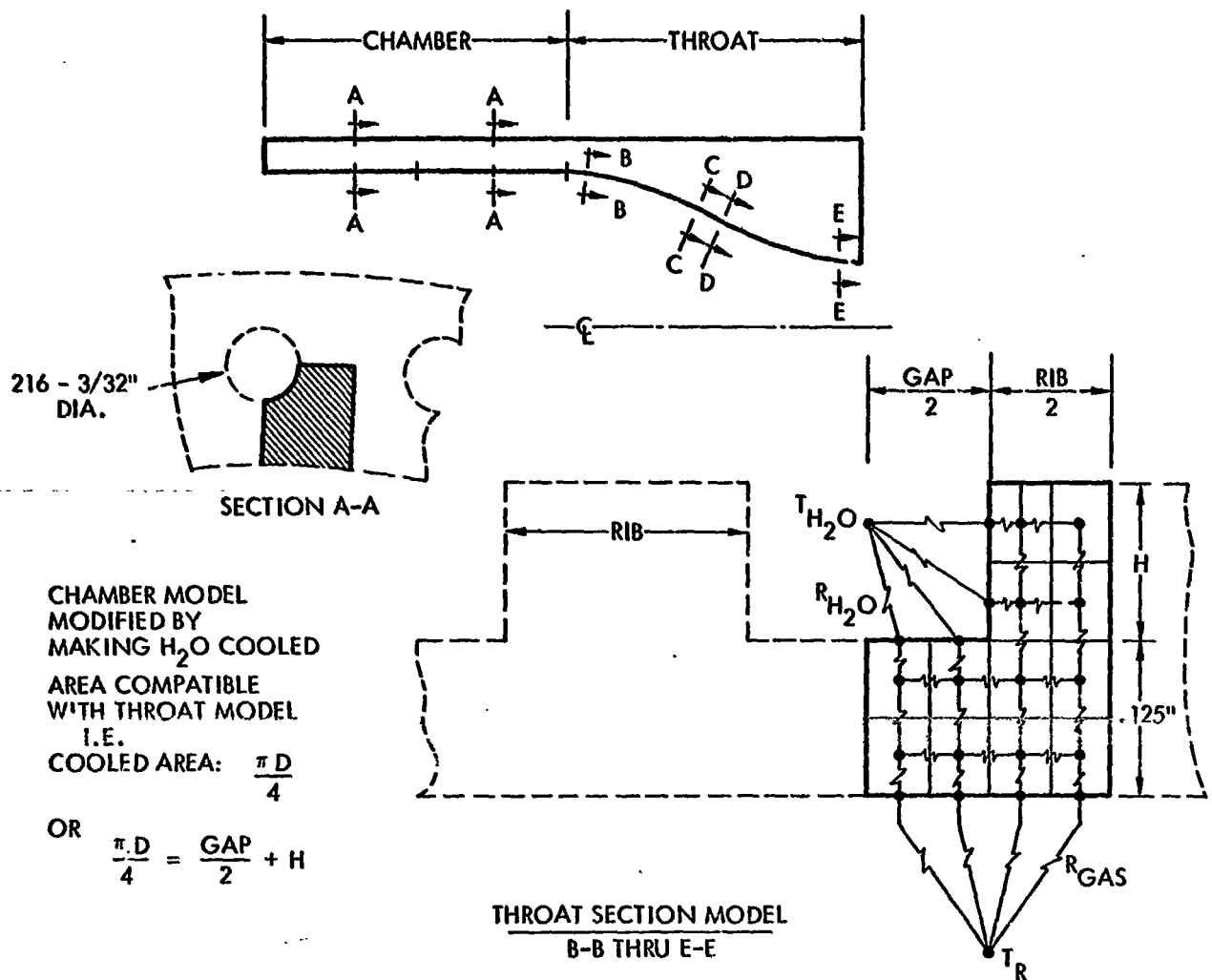
FIGURE 3-61 COOLANT WATER VELOCITY

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Wall Operating Temperatures

The anticipated wall temperatures along each chamber section and through the throat were determined for the maximum anticipated heat loads, the passage designs shown, and the selected water flow rates. The thermal model and the wall locations examined are shown in Figure 3-62. Each location depicts definite passage dimensions and should quite accurately represent the hardware designs. The steady-state wall temperatures resulting from the analysis are shown in Figure 3-63 for all chamber and throat locations.

The throat analysis showed a maximum wall temperature of 570°F for the worst possible heat flux condition. The other wall points along the throat section showed temperatures varying down to 450°F. The cylindrical chamber sections each showed a maximum wall temperature of 450°F. The mean and cold wall temperatures were also shown in Figure 3-63 to facilitate thermal growth and stress calculations for the final design.



(DIMENSIONS IN INCHES)

SECTION	LOCATION	GAP	RIB	H
A-A	CHAMBER, CYLINDRICAL	.074	.102	.037
B-B	INLET TO THROAT SECTION	.155	.137	.125
C-C	UPSTREAM OF TRANSITION	.155	.086	.126
D-D	DOWNSTREAM OF TRANSITION	.100	.140	.125
E-E	THROAT PLANE	.100	.086	.125

FIGURE 3-62 ALUMINUM CHAMBER THERMAL MODELS

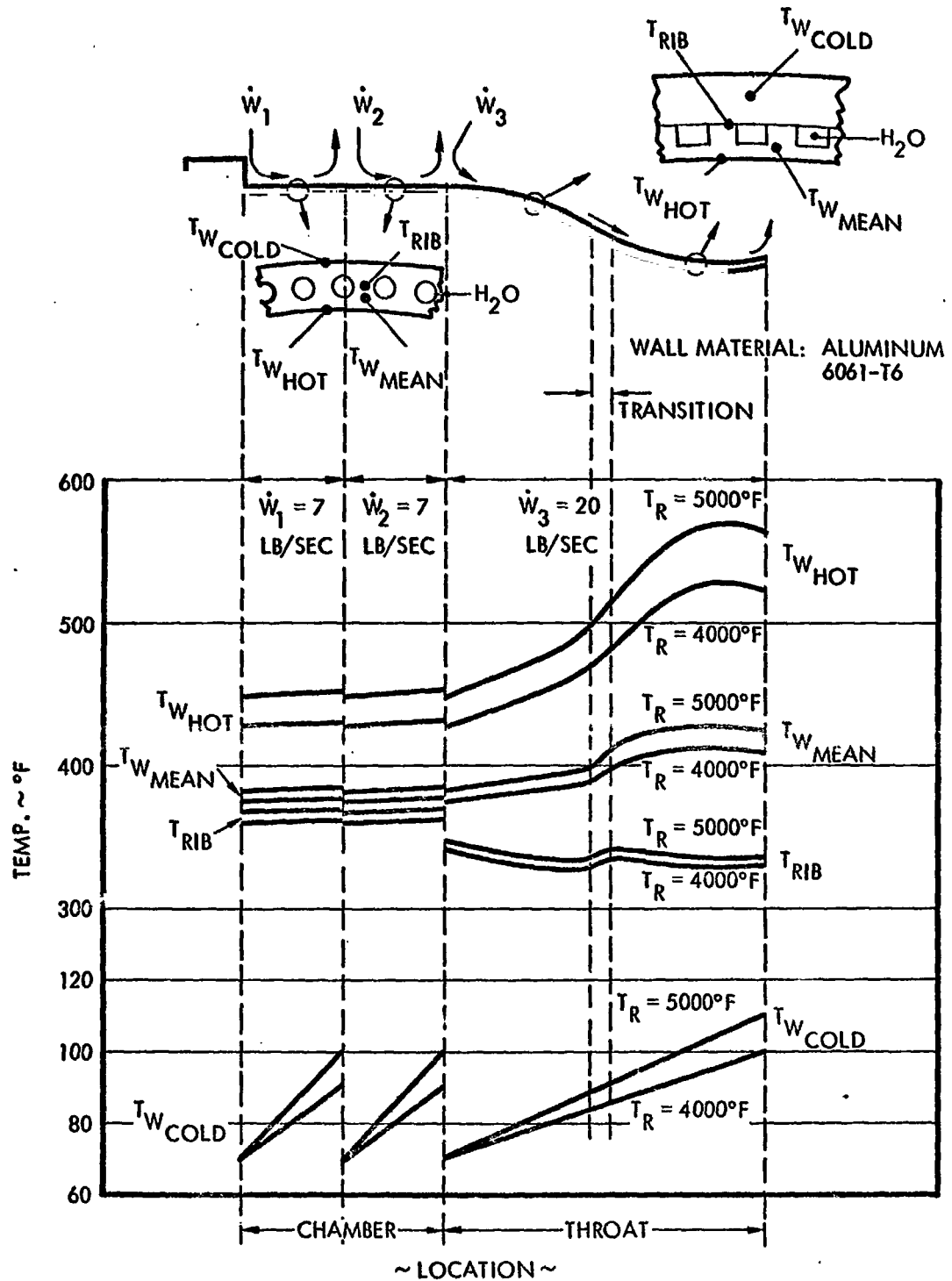


FIGURE 3-63 STEADY-STATE WALL TEMPERATURES

3.6.3 Aluminum Chamber Data Reduction Program

A data reduction technique was developed for use with the Qual B aluminum water-cooled chamber. The data reduction program (WATCH) was written for the SDS 930 computer located at CTS.

- A typical flow schematic is shown in Figure 3-64. Most of the parameter designations are depicted at specific locations in the flow scheme.
- Figure 3-65 shows the total water flow schematic for the aluminum test rig.

Measured Parameters

TIN	Water temperature at main inlet header		
TE(T)	Water temperature at main discharge header		
W(T)	Total flow to main inlet header		
W(A)	Flow to Section A		
W(B)	Flow to Section B		
W(C)	Flow to Section C		
TE	Water temperature in each exit line (segment discharge)		
TE(A)	section A	(17 Total)	
TE(B)	section B	(17 Total)	
TE(C)	section C	(28 Total)	
		62 Total Chamber	

Calculated Parameters

DELT	Temperature rise through each segment		
W(TT)	Total flow [check on W(T)]		
W	Flow through each segment		
V	Velocity through each slot or segment		
QA	Average heat flux for each segment		
TWH	Average hot side wall temperature (approximate)		

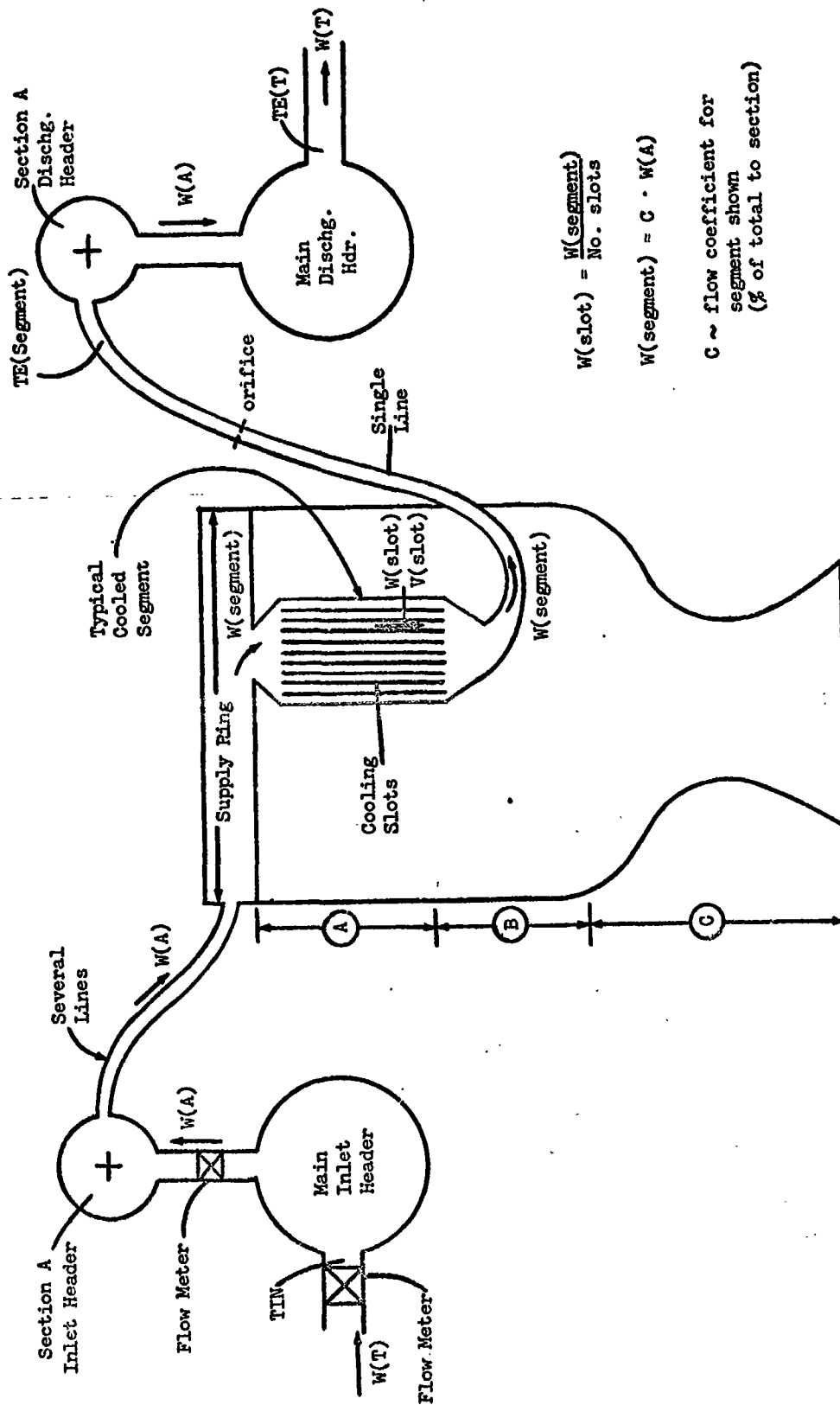


FIGURE 3-64 TYPICAL FLOW SCHEMATIC

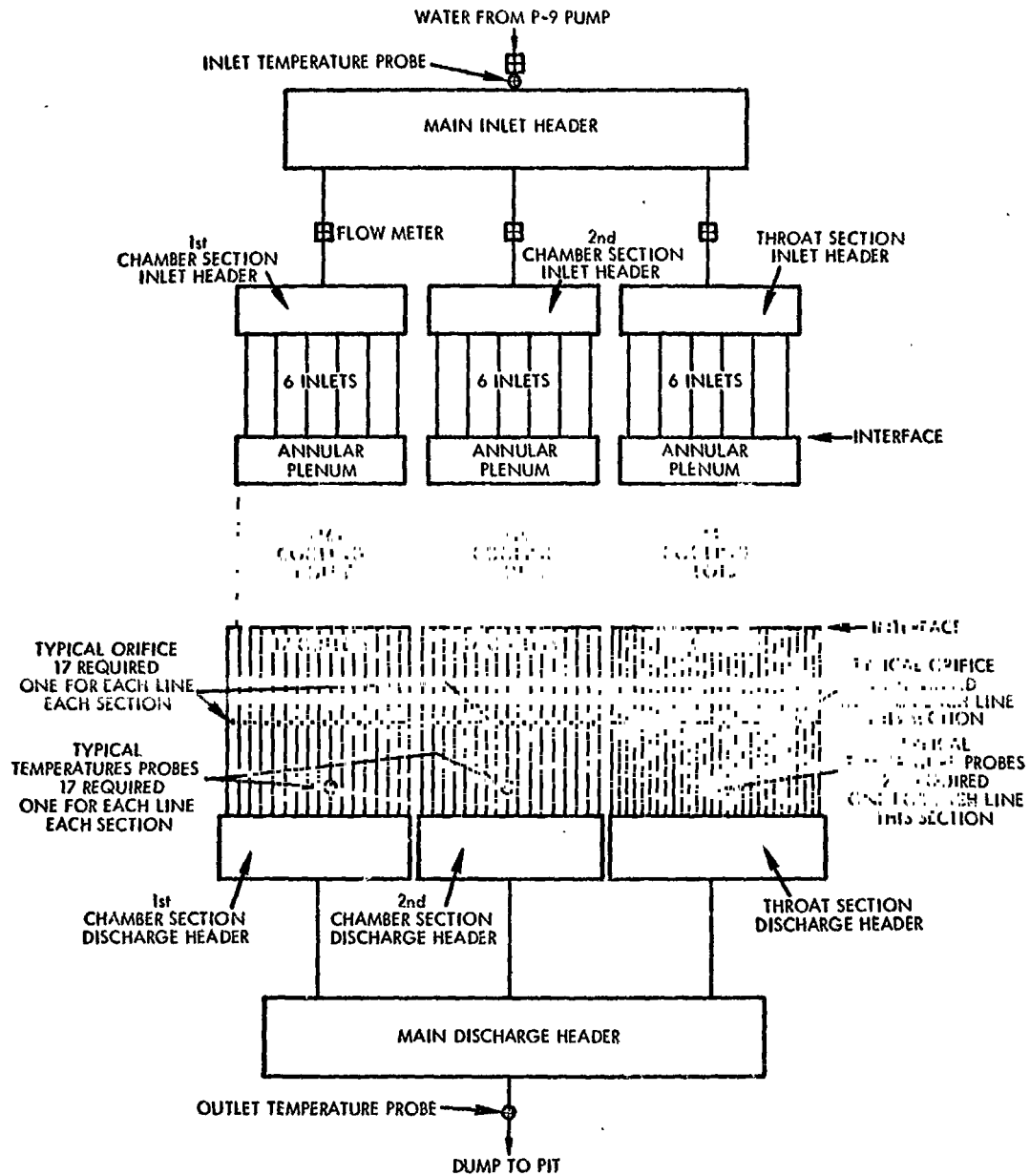


FIGURE 3-65 TOTAL WATER FLOW SCHEMATIC

Input Parameters

CF	Segment flow coefficient	d'less
NS	Number of cooling slots per segment	d'less
AR	Segment heat transfer area	in ²
K	Constant in velocity calculation	lb _m /ft

Calculations

DEL Δ = TE - TIN (average over a selected time increment)

$$W = CF \cdot W(n) \text{ where } n = A, B, \text{ or } C$$

$$W(TT) = W(A) + W(B) + W(C)$$

$$V = \frac{W}{K \cdot NS}$$

$$QA = \frac{W \cdot DEL\Delta}{AR}$$

$$TWH = 350 + 63 \cdot QA$$

The complete aluminum chamber assembly was water-flowed resulting in the flow coefficients listed in Table 3-7. These calibration coefficients were subsequently used in the WATCH program as constant input items. Complete documentation and listings of the WATCH program are provided in Reference 28.

TABLE 3-7 COLD FLOW CALIBRATION

1st Chamber Section			2nd Chamber Section			Throat		
Segment	Flow Coef.	No. Cooling Slots	Segment	Flow Coef.	No. Cooling Slots	Segment	Flow Coef.	No. Cooling Slots
001-2.5	0.0839	18	002-2.5	0.0840	18	Th-10	0.0421	6
-32.5	0.0830	18	-32.5	0.0852	18	-25	0.0424	6
-62.5	0.0839	18	-62.5	0.0838	18	-40	0.0424	6
-92.5	0.0845	18	-92.5	0.0852	18	-55	0.0424	6
-110	0.0132	3	-110	0.0129	3	-70	0.0424	6
-115	0.0130	3	-115	0.0131	3	-85	0.0412	6
-120	0.0132	3	-120	0.0131	3	-100	0.0412	6
-125	0.0131	3	-125	0.0129	3	-110	0.0135	2
-130	0.0129	3	-130	0.0129	3	-115	0.0141	2
-135	0.0130	3	-135	0.0130	3	-120	0.0136	2
-152.5	0.0839	18	-152.5	0.0830	18	-125	0.0135	2
-182.5	0.0832	18	-182.5	0.0824	18	-130	0.0135	2
-212.5	0.0842	18	-212.5	0.0824	18	-135	0.0135	2
-242.5	0.0853	18	-242.5	0.0824	18	-145	0.0412	6
-272.5	0.0840	18	-272.5	0.0835	18	-160	0.0412	6
-302.5	0.0828	18	-302.5	0.0872	18	-175	0.0413	6
-332.5	0.0828	18	-332.5	0.0828	18	-190	0.0417	6
						-205	0.0414	6
						-220	0.0415	6
						-235	0.0412	6
						-250	0.0420	6
						-265	0.0415	6
						-280	0.0417	6
						-295	0.0419	6
						-310	0.0423	6
						-325	0.0419	6
						-340	0.0417	6
						-355	0.0419	6

3.6.4 Qual B Baseline Thermal Environment Characterization

Tests were performed on the LMDE aluminum water cooled chamber and the resultant data were reduced in support of the LMDE Qual B baseline thermal environment characterization program. The engine was subjected to a test series consisting of seven thrust levels covering the entire thrust range for the LMDE chamber. These tests included steady state dwells at 10%, 25%, 40%, 50%, 60%, 65% and 100% thrust levels.

This section presents the axial and circumferential heat fluxes in the engine as a function of thrust level as well as averaged heat fluxes in the throat and chamber as a function of thrust level.

3.6.4.1 Conclusion and Pertinent Information

Emphasis on heat flux data has been placed on test VA2-640 due to the quality and duration of the steady-state data acquired. Analysis of this test has yielded the following significant data and information.

- The water flows to each section were nominally as follows

Section A - 5.36 lb/sec

Section B - 5.42 lb/sec

Section C - 13.63 lb/sec

- The minimum segment water velocity of 7.9 ft/sec occurred in the test rig in Section A.
- The water temperature rise at 10% thrust was only 3.5 to 7.0 degrees for all the water segments. Since the deviation in temperature data is as much as 1°F the heat flux accuracy at this thrust level cannot be considered to be sufficiently reliable.
- Section A (nearest head end) showed a higher average heat flux than sections B or C for all thrust levels between 10% and 65%.
- Section B showed a higher average heat flux than Section C for thrust levels between 10% and 40%.
- Section C (throat region) showed the highest average heat flux only at FTP (when compared with sections A and B).

- Approximate average heat flux at each section are as follows:

Thrust Level %	Q/A (Btu/in ² -sec)		
	Section A	Section B	Section C
10	0.29	0.24	0.19
25	0.71	0.67	0.57
40	1.09	0.89	0.84
50	1.26	1.04	1.06
60	1.46	1.24	1.36
65	1.52	1.29	1.43
FTP	1.21	1.36	1.88

- All hot side wall temperatures were estimated to be between 360 and 470°F.

3.6.4.2 Heat Flux Summary

The aluminum water cooled chamber test rig was designed and constructed to provide for the measurement of heat transfer to local areas of the chamber walls during steady state engine operation. Determination of heat flux, both axially and circumferentially, was made by measuring water flow rate and temperature rise in the appropriate segments. Figure 3-41 shows the chamber and throat sections indicating the location of the various water outlets.

Design criteria for this test rig were presented in Reference 8 and investigations of the thermal and hydraulic adequacy of the design are presented in Section 3.6.2 and Reference 8.

Tests were conducted on April 28, 1969 using HEA 023 on the TRW VETS facility. Three tests were performed as follows:

- VA2 - 639 Calibration test
- VA2 - 640 Complete run through with steady state dwells for 25 secs at 10%, 25%, 40%, 50%, 60%, 65% and 100% thrust levels. The duty cycle for this test is illustrated in Figure 3-66.

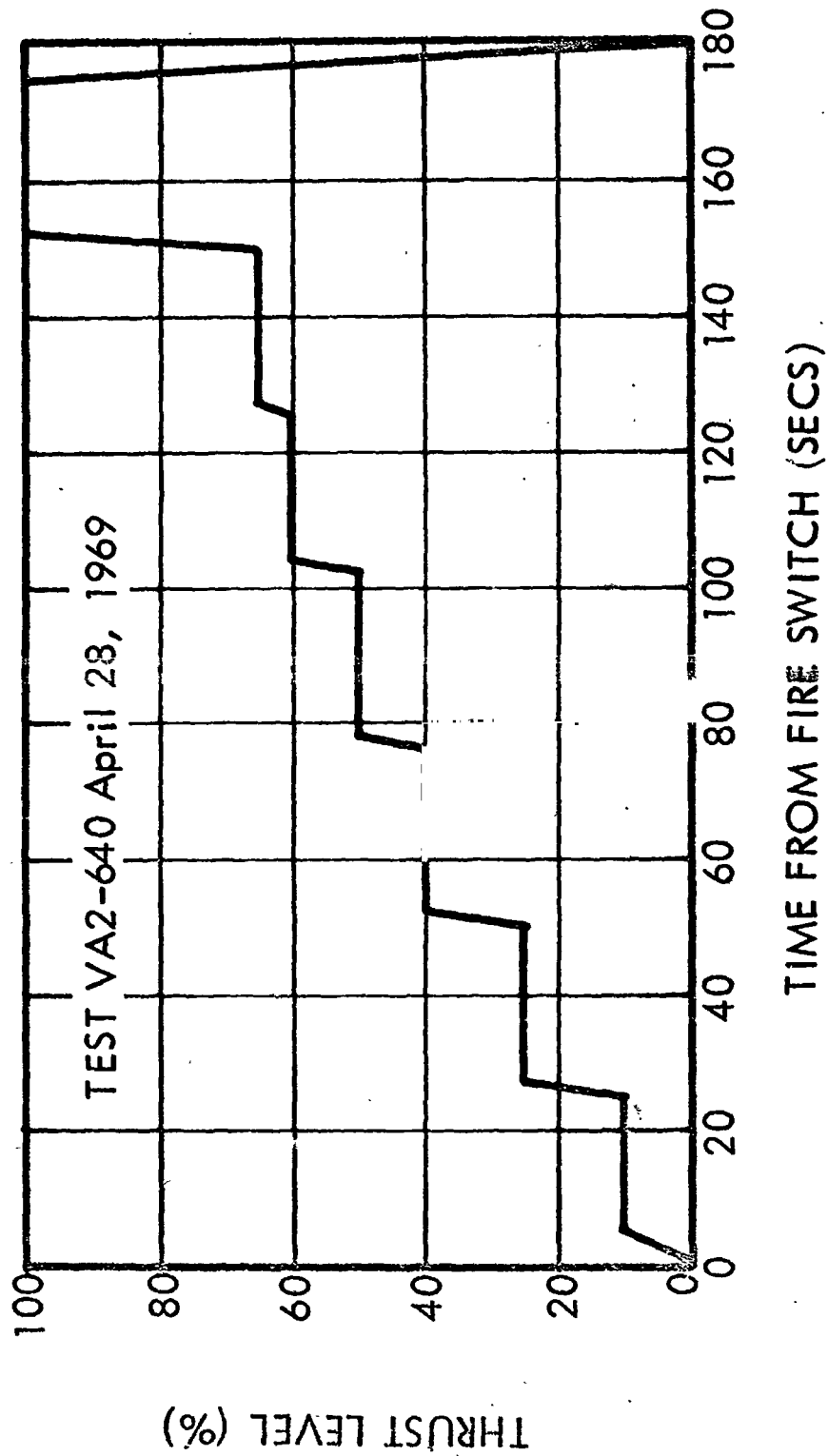


FIGURE 3-66 TEST FIRING DUTY CYCLE

- VA2 - 641 Partial run-through with steady state dwells for 20 secs at 10%, 25%, 40%, 50%, and 60%.

Only the data from VA2 - 640 was studied in any detail since test VA2 - 641 produced, essentially, redundant data. A quick look at the reduced data for the thrust levels which were performed on VA2 - 641 revealed results which were identical to that for VA2 - 640.

Typical strip chart temperature traces for each section exit water are shown in Figures 3-67, 3-68, and 3-69.

The heat flux as a function of angular location for different thrust levels and axial locations are presented in Figures 3-70, 3-71, and 3-72.

The heat fluxes and wall temperatures have been averaged for each station. These average heat fluxes and wall temperatures are tabulated below in Tables 3-8 and 3-9 and shown graphically in Figures 3-73 through 3-76.

TABLE 3-8 AVERAGE HEAT FLUX AT EACH STATION

Thrust Level (%)	Average Mixture Ratio	Average Q/A (Btu/in ² -sec)		
		Station A	Station B	Station C
10	1.544	.295	.243	.188
25	1.569	.710	.673	.572
40	1.583	1.091	.896	.840
50	1.588	1.257	1.038	1.064
60	1.602	1.457	1.242	1.359
65	1.595	1.523	1.293	1.430
FTP	1.581	1.213	1.357	1.883

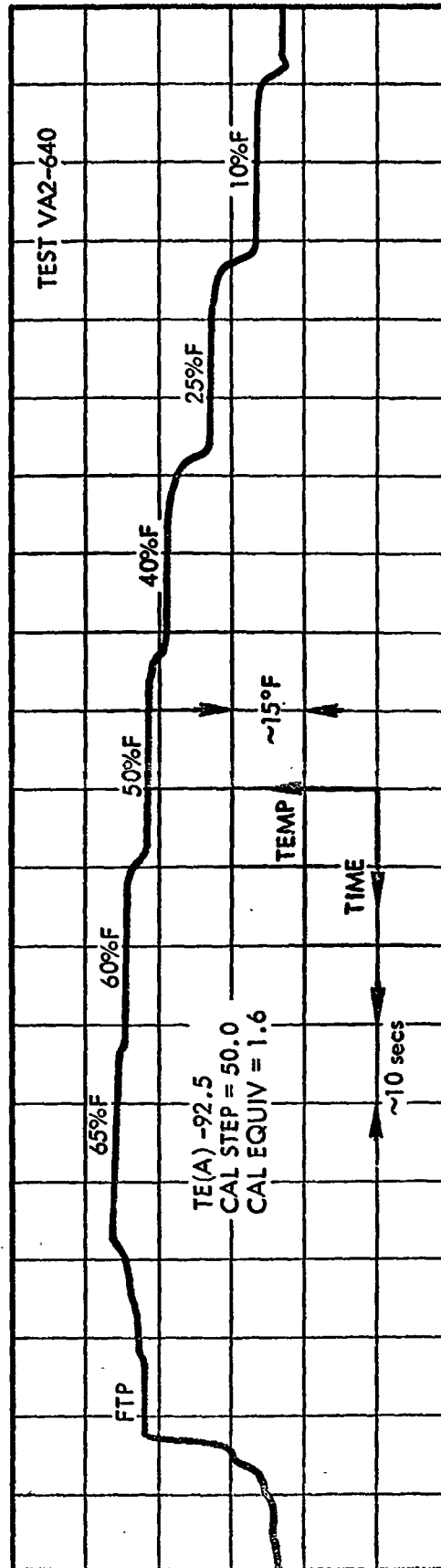


FIGURE 3-67 STRIP CHART TEMPERATURE TRACE, SECTION A

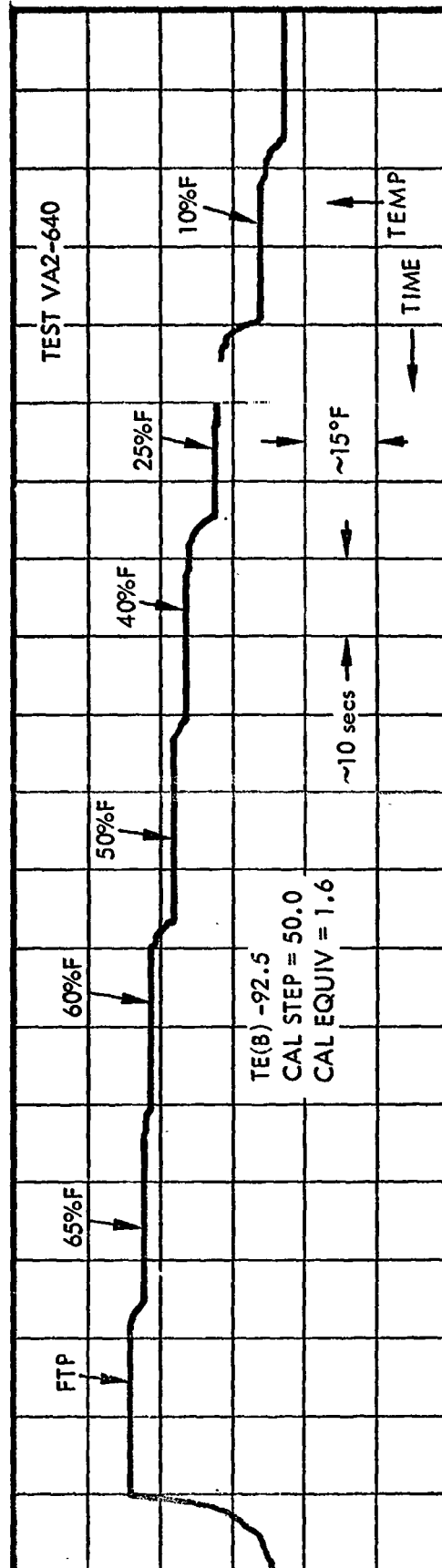


FIGURE 3-68 STRIP CHART TEMPERATURE TRACE, SECTION B

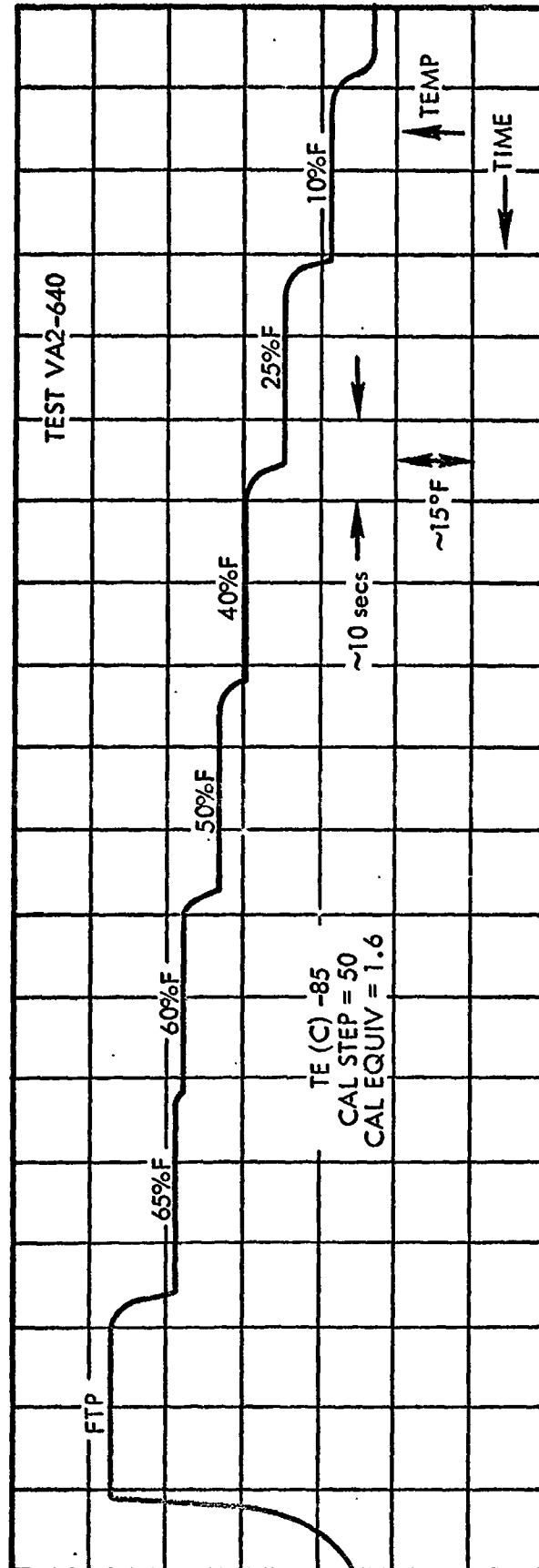


FIGURE 3-69 STRIP CHART TEMPERATURE TRACE, SECTION C

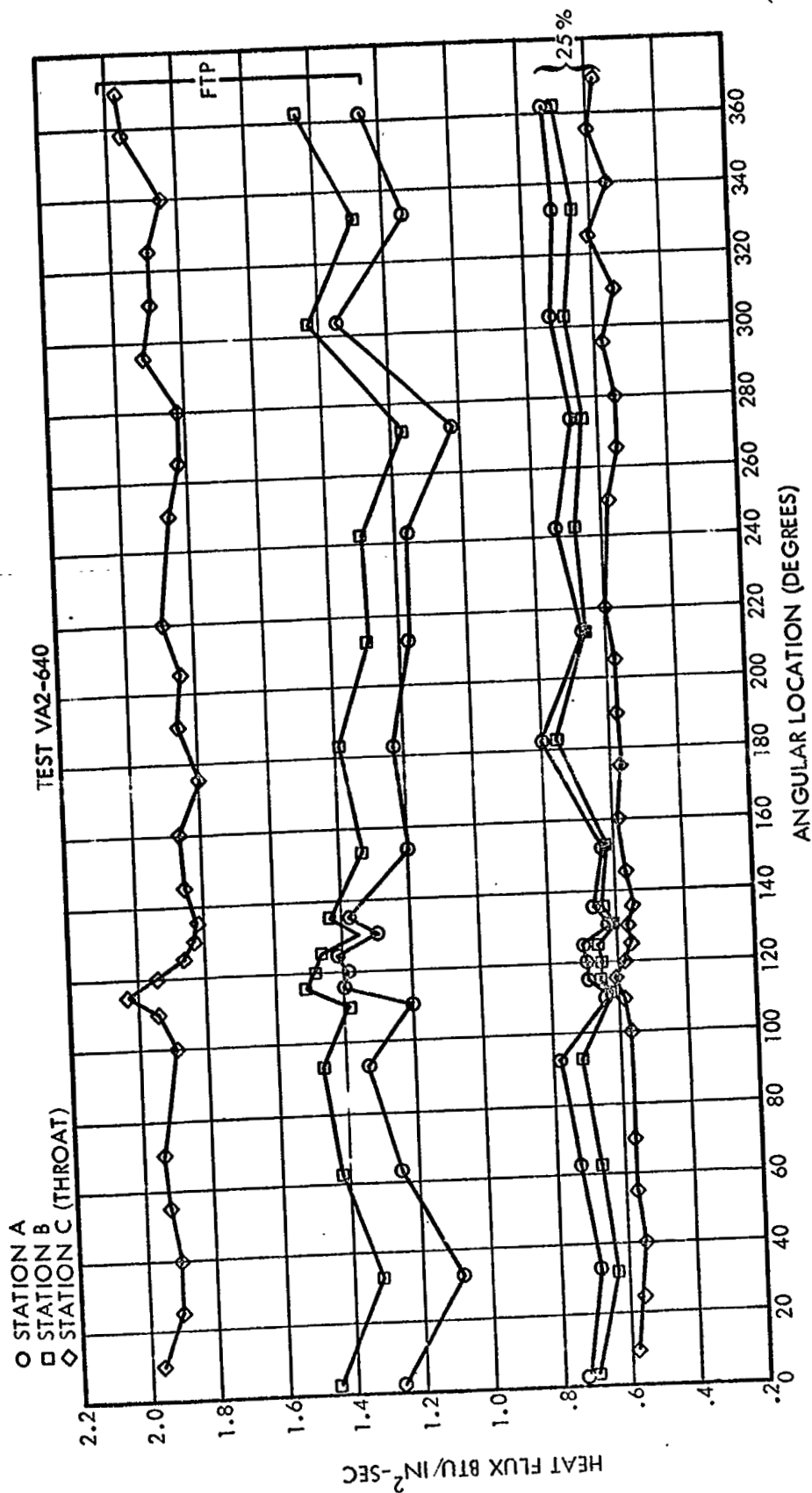


FIGURE 3-70 ALUMINUM CHAMBER HEAT FLUX AT FTP AND 25% THRUST

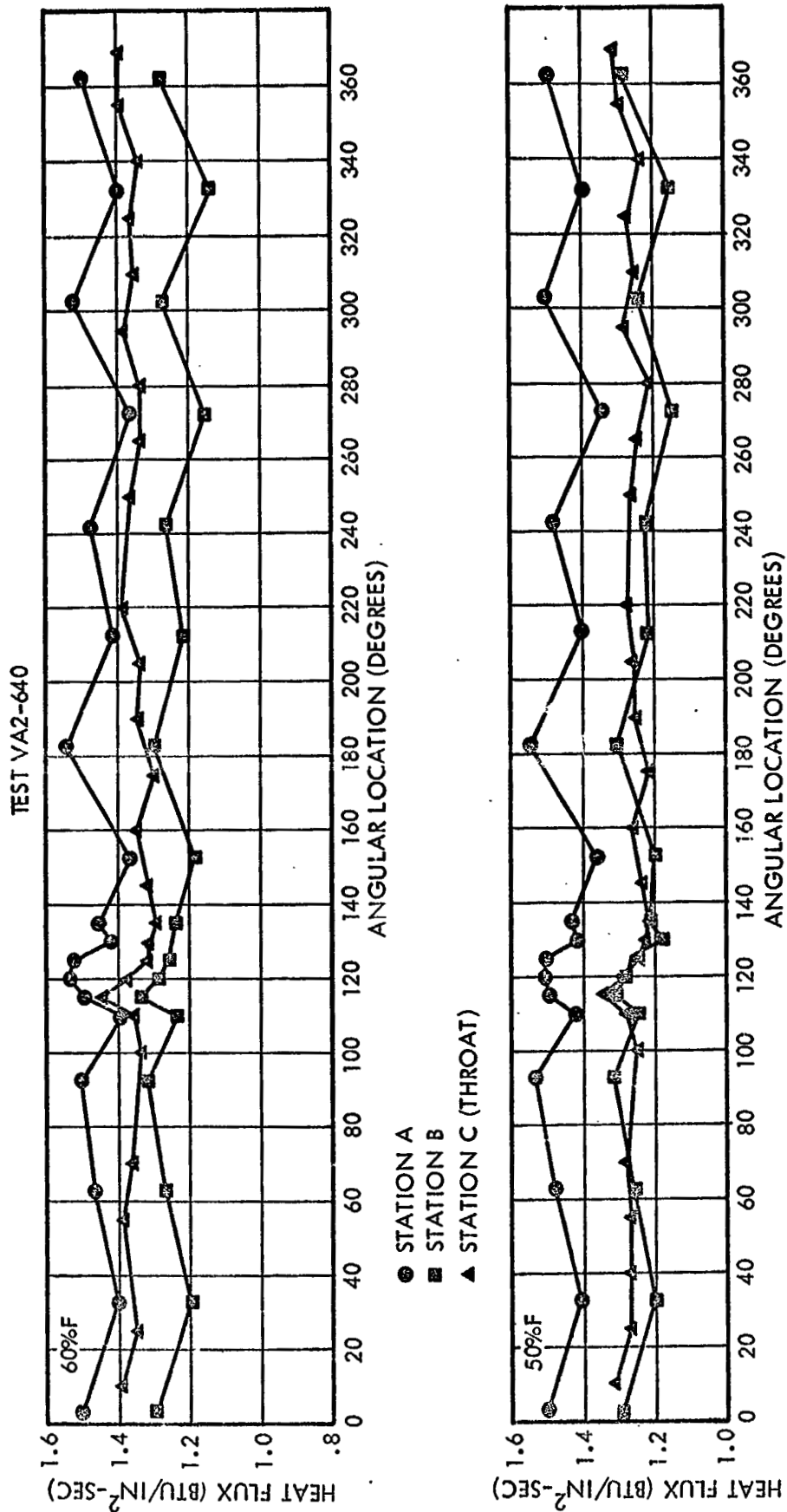


FIGURE 3-71 ALUMINUM CHAMBER HEAT FLUX AT 50% AND 60% THRUST

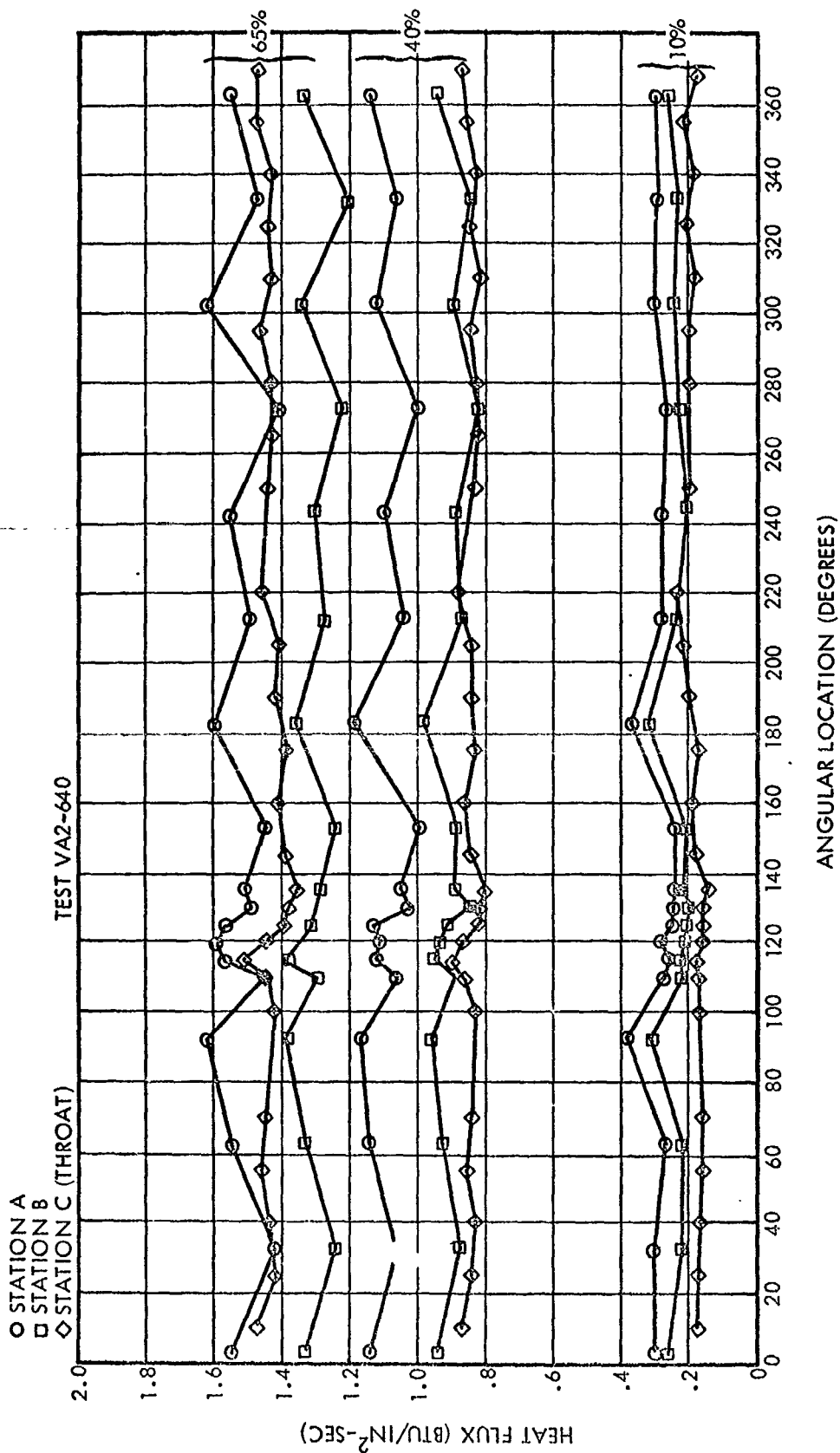


FIGURE 3-72 ALUMINUM CHAMBER HEAT FLUX AT 10%, 40%, AND 65% THRUST

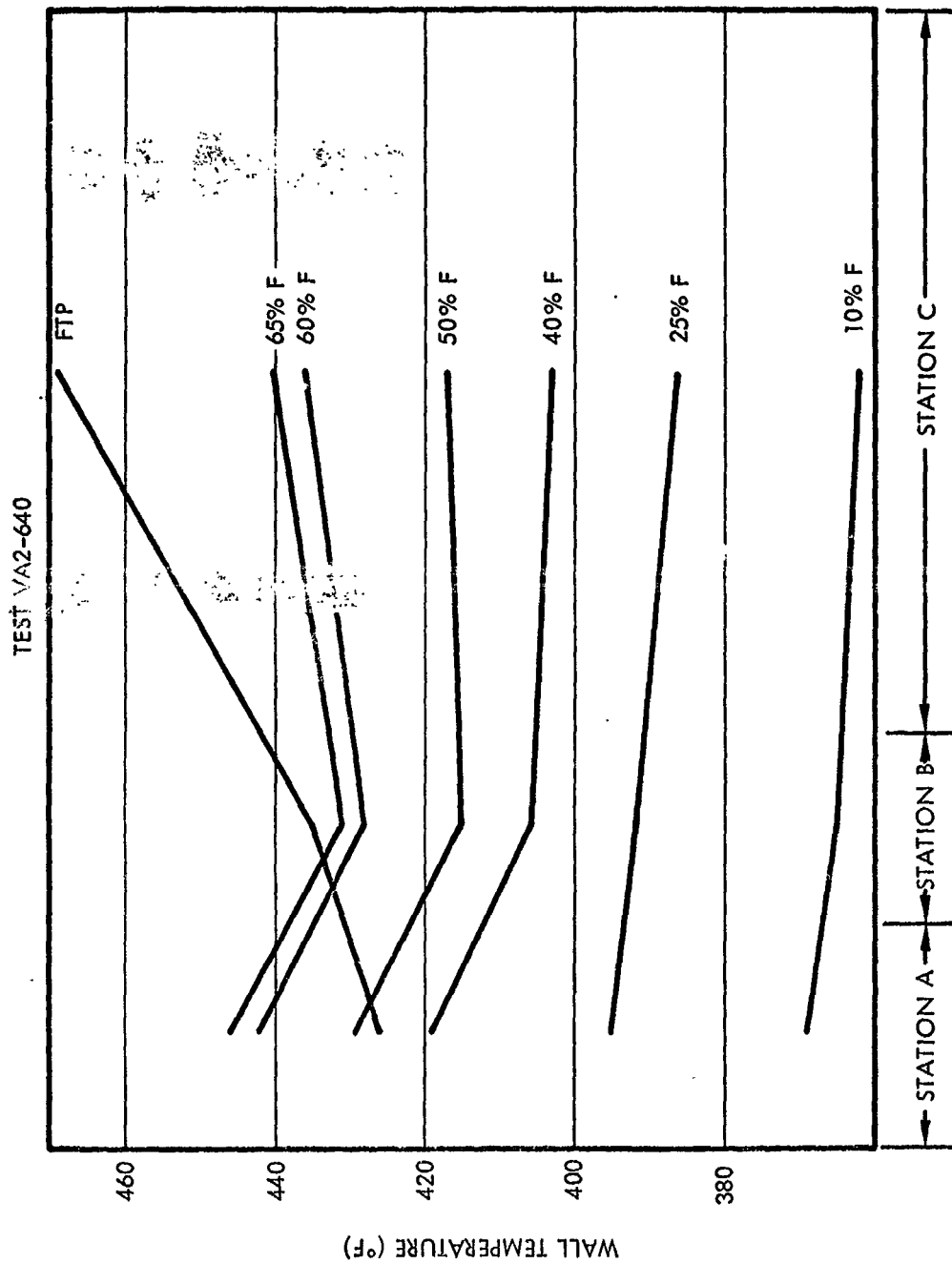


FIGURE 3-73 AVERAGE WALL TEMPERATURE PROFILE

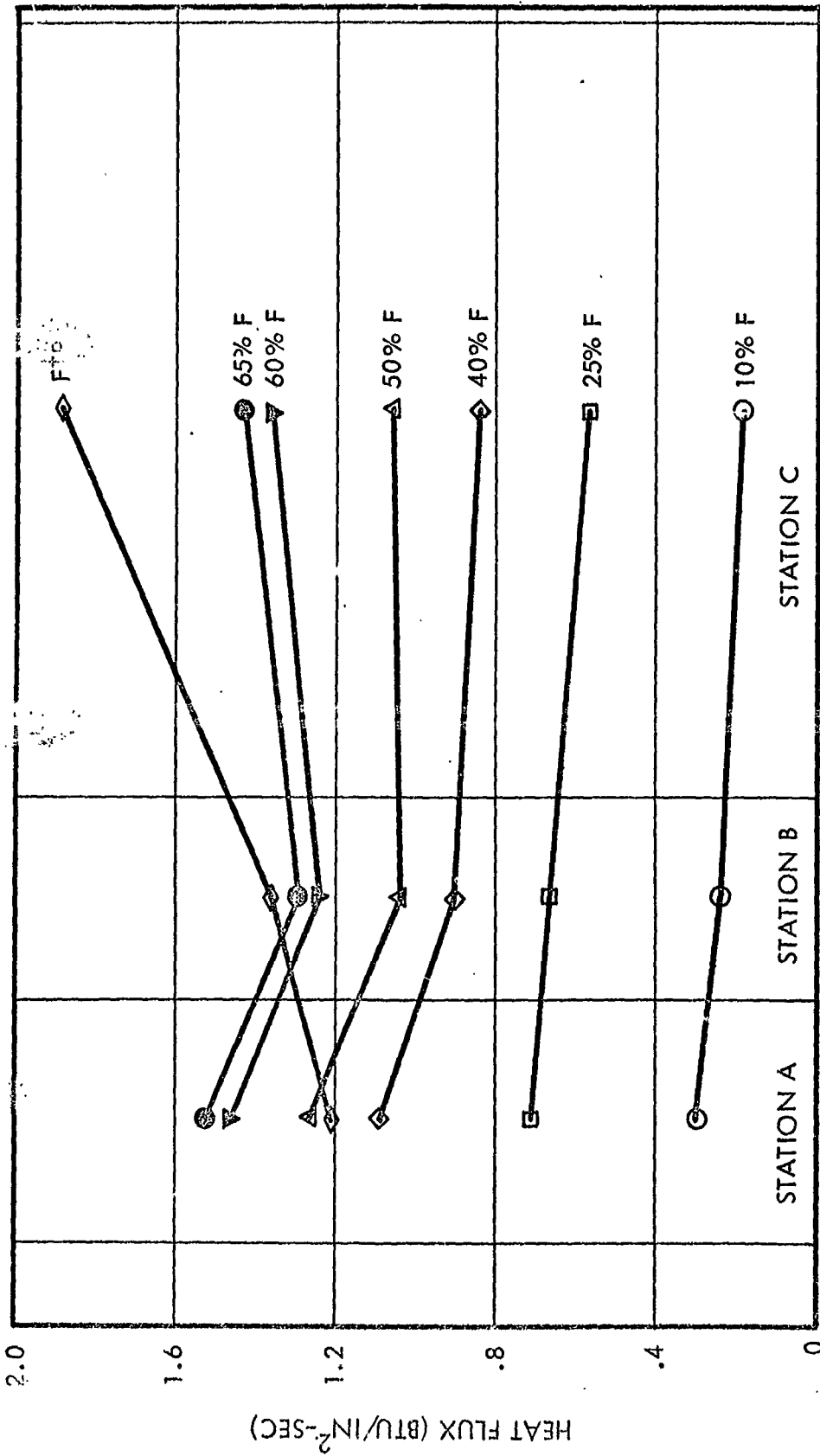


FIGURE 3-74 AVERAGE HEAT FLUX PROFILE

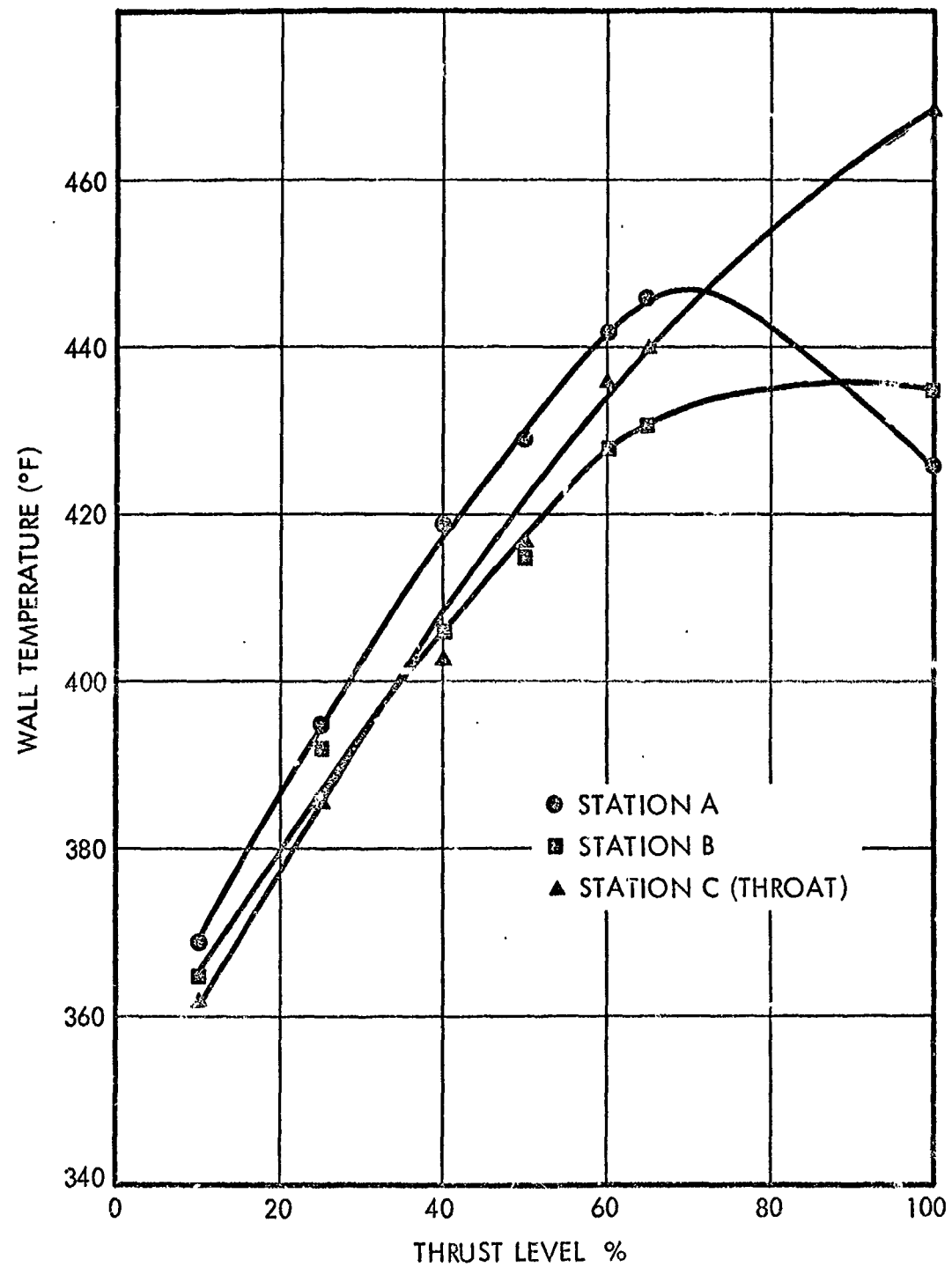


FIGURE 3-75 WALL TEMPERATURE VARIATION

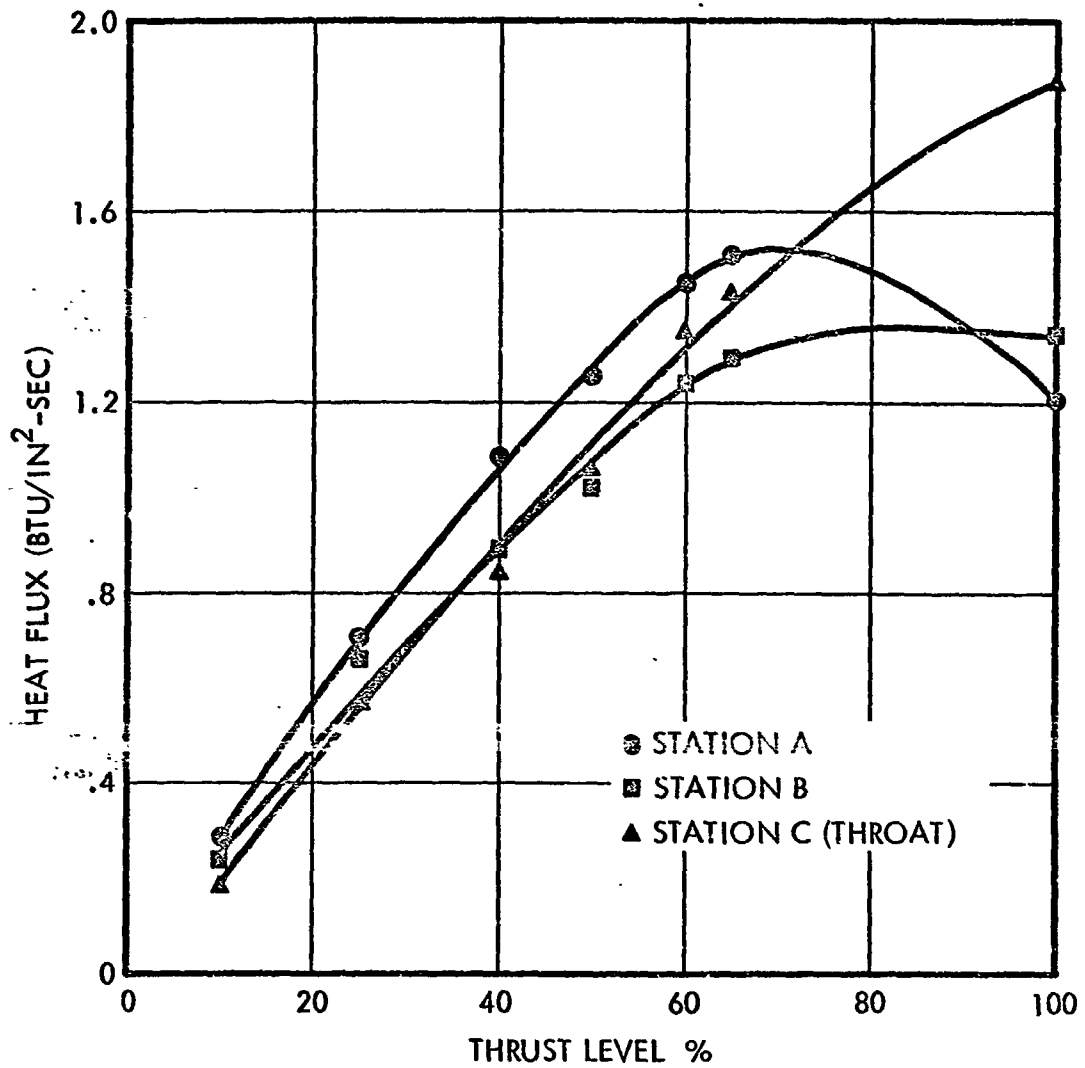


FIGURE 3-76 HEAT FLUX VARIATION

TABLE 3-9 AVERAGE WALL TEMPERATURE AT EACH STATION

Thrust Level (%)	Wall Temperature ($^{\circ}\text{F}$)		
	Station A	Station B	Station C
10	369	365	362
25	395	392	386
40	419	406	403
50	429	415	417
60	442	428	436
65	446	431	440
FTP	426	435	469

Discussion of the Results

The fact that higher heat fluxes are being recorded in the chamber than in the throat was a cause of some concern as it was not in accord with data from previous tests. At present there is no exact explanation for the cause of these results; however, several mechanisms are hypothesized to be contributing to the effect.

- The fact that this phenomenon has not shown up in previous tests indicates that the low wall temperature in the chamber for this series of tests may be a significant contributing factor.
- Tests on other engines with similar configurations have revealed that during the early part of the firing, when the wall is cool, the heat transfer coefficient in the chamber is very high, in fact, many times higher than the theoretical Bartz coefficient. Although the recovery temperature in this region is low, when coupled with the very high heat transfer coefficient the result is a high heat flux to the wall.
- The high chamber heat transfer coefficient may be due to the manner in which the barrier coolant film lies along the chamber walls in either a thin liquid layer next to the wall or a two-phase outer zone.

It is also possible that the high coefficient is a result of oxidizer impingement on the wall. This possibility has been essentially substantiated from streak chamber tests at 25% thrust. The chamber cylindrical section was found to be highly eroded while the throat showed very little erosion (test VA1-665).

3.7 COMPARISON OF HEAT SINK AND WATER-COOLED CHAMBERS

An evaluation and comparison study was made of the thermal results from tests of the three basic Qual test rigs.

- Steel water-cooled chamber
- Steel heat sink chamber
- Aluminum water-cooled chamber

All of the reduced heat flux data has been presented previously.

Throat Heat Flux

Average heat fluxes in the throat region are tabulated below for the three series of tests performed on the heat sink, aluminum water-cooled chamber and the steel water-cooled chamber. The heat sink data has been extrapolated (assuming a linear relationship) to the same wall temperature as the aluminum chamber wall temperature.

Thrust Level %	Wall Temperature (°F)		Average Throat Heat Flux (Btu/in ² -sec)		
	Aluminum W/C and Heat Sink	Steel W/C	Heat Sink (Throat Integrated Average)	Aluminum Water Cooled	Steel Water Cooled
10	362	410	N.A.	0.188	0.230
25	386	570	0.705	0.572	0.845
40	403	645	(No data)	0.840	1.125
50	417	680	1.180	1.064	1.265
60	436	(No data)	(No data)	1.359	(No data)
65	440	765	1.245	1.430	1.596
FTP	469	870	2.000	1.883	1.980

Figure 3-77 shows the throat heat flux as a function of thrust level for each of the three configurations tested. The heat flux data for the heat sink engine has been extrapolated back to the same wall temperature as that for the aluminum water cooled chamber. This extrapolation is questionable since the curve of heat flux versus wall temperature for

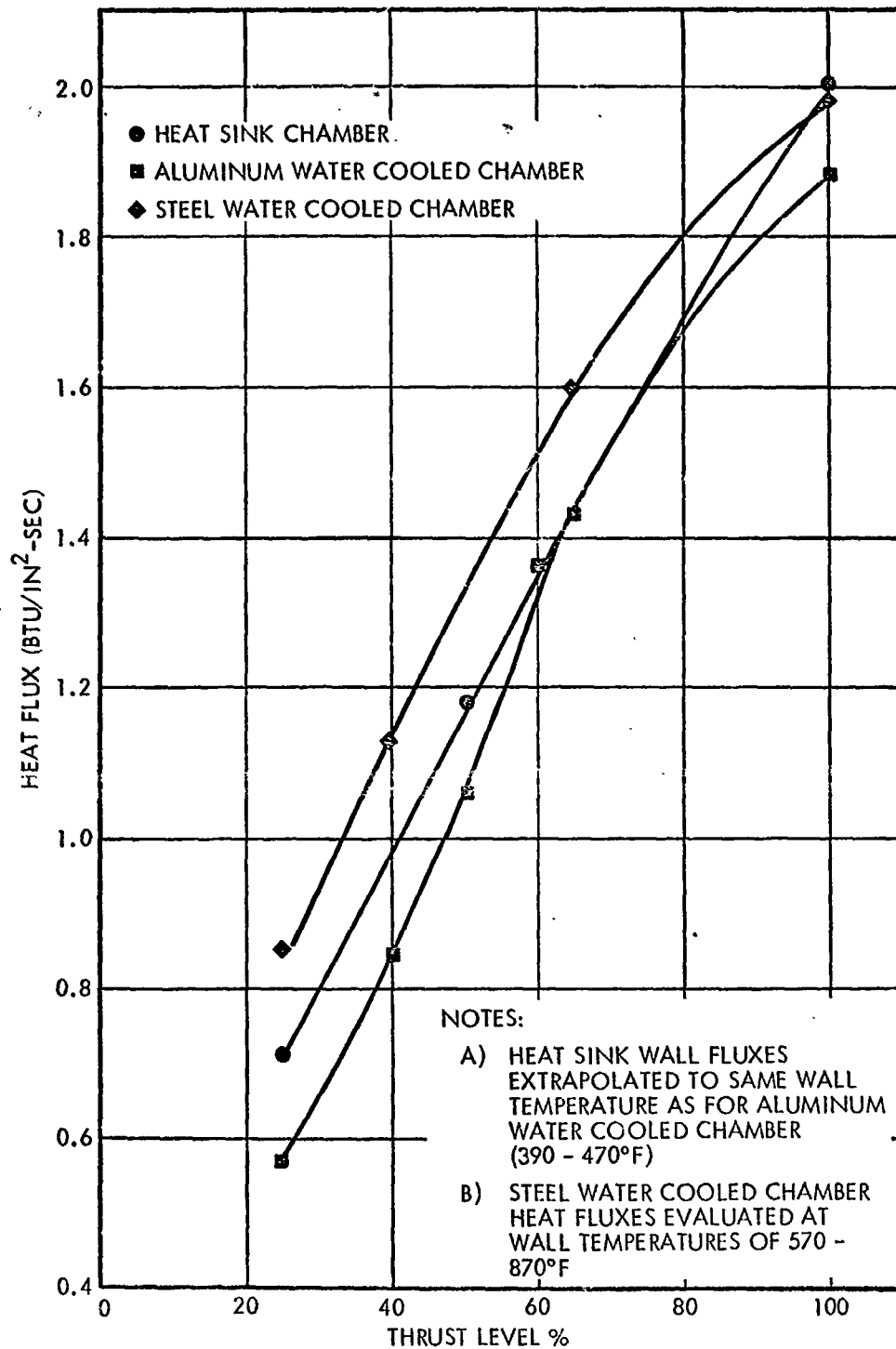


FIGURE 3-77 THROAT HEAT FLUX COMPARISONS

the heat sink engine is, in some cases, not clearly enough defined (with respect to linearity) to permit accurate extrapolation. This is particularly true of the data for the 65% thrust level and explains why the 65% thrust point does not fall on a smooth curve.

At thrust levels greater than about 50% the agreement between the three sets of data is fairly good. At thrust levels below 50% the differences between the three sets of data are more marked.

One obvious factor which may be contributing to the differences in the three sets of data is mixture ratio variation. Previous studies have indicated that fairly large off-nominal mixture ratio variations theoretically result in relatively insignificant changes in the Bartz heat transfer coefficient at the throat. Also, studies of combustion temperature as a function of mixture ratio have indicated that small changes in mixture ratio will have little effect on the combustion temperature. However, small changes in mixture ratio may have a significant effect upon the oxidizer/fuel injector spray distribution. There may be a correspondingly marked effect upon the interaction between this spray and the barrier coolant film, and hence upon the heat flux.

There is very little experimental evidence which reveals the effect of mixture ratio changes upon throat heat fluxes for this engine. Two sets of tests were conducted on the heat sink configuration at 25% thrust with the only apparent difference between the two being the mixture ratio. Throat heat flux data was reduced and extrapolated to a wall temperature of 386°F and the results are presented below.

<u>Test No.</u>	<u>Date</u>	<u>Mixture Ratio</u>	<u>Throat Heat Flux (Btu/in²-sec)</u>
VA1-688	1/17/69	1.60	0.513
VA1-692	1/28/69	1.63	0.705

Below are tabulated the mixture ratios for the tests performed upon the three different configurations:

Thrust Level %	Overall Mixture Ratio		
	Heat Sink	Aluminum W/C	Steel W/C
10	N.A.	1.54	1.61
25	1.63	1.57	1.59
40	(No test)	1.58	1.60
50	1.61	1.59	1.60
60	(No test)	1.60	(No test)
65	1.61	1.59	1.58
FTP	1.61	1.58	1.60

Chamber Heat Flux

Average heat fluxes in the cylindrical chamber section closest to the throat region are tabulated below for the three series of tests. The heat sink data has been extrapolated to the same wall temperatures as that of the aluminum water-cooled chamber.

Thrust Level %	Average Wall Temperature ($^{\circ}\text{F}$)		Average Chamber Heat Flux ($\text{Btu/in}^2\text{-sec}$)		
	Aluminum W/C and Heat Sink	Steel W/C	Heat Sink (Station 3)	Aluminum Water Cooled	Steel Water Cooled
10	365	397	N.A.	0.243	0.182
25	392	490	0.415	0.673	0.547
40	406	575	(No data)	0.890	0.850
50	415	613	0.628	1.038	1.000
60	428	(No data)	(No data)	1.242	(No data)
65	431	660	0.749	1.293	1.190
FTP	435	697	0.883	1.397	1.335

Figure 3-78 shows the heat flux at a chamber location just upstream of the throat as a function of the thrust level for each of the three configurations tested. The heat flux data from the heat sink engine has been extrapolated back to approximately the same wall temperature as that for the aluminum water-cooled chamber.

The data for the two water-cooled engines shows close agreement. The fact that the aluminum water-cooled chamber shows a slightly higher heat flux, when compared with the steel water-cooled chamber, is probably due to lower wall temperatures (higher driving potential).

The heat sink data shows chamber heat fluxes significantly lower than for the other two configurations. The higher heat flux for the two water-cooled configurations is probably due to the altered thermal environment in the vicinity of the walls due to the lower wall temperature. This phenomenon has been previously discussed in Section 3.6.4.

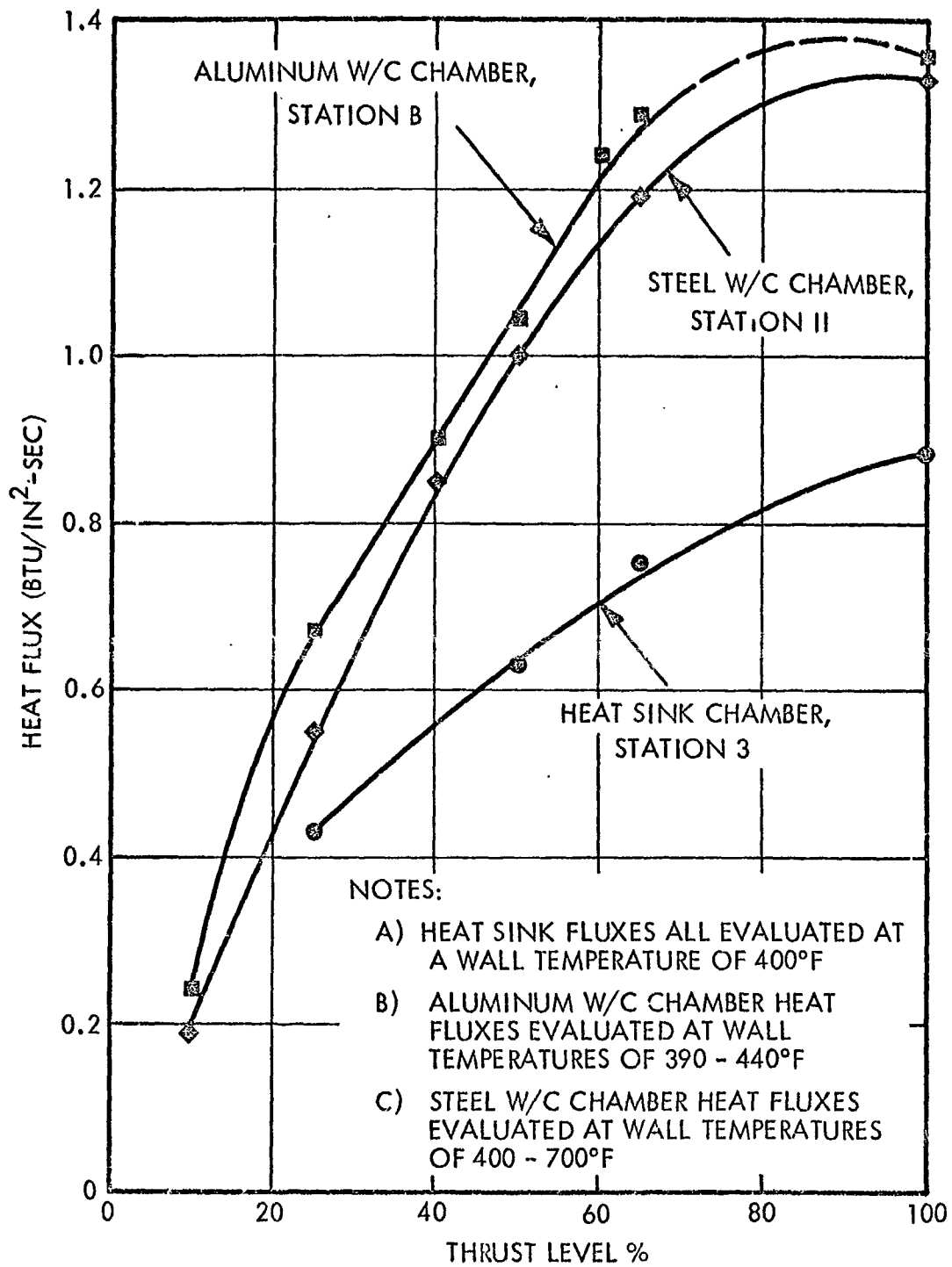


FIGURE 3-78 CHAMBER HEAT FLUX COMPARISONS

3.8 THERMAL ENVIRONMENT/EROSION CORRELATION

This documentation presents the thermal environment-erosion data correlations obtained for both the Qual B baseline and the improved flow distribution test series. The experimental heat transfer studies were conducted with specially designed and instrumented throat sections to obtain the wall thermal environment data.

3.8.1 Qual B Baseline

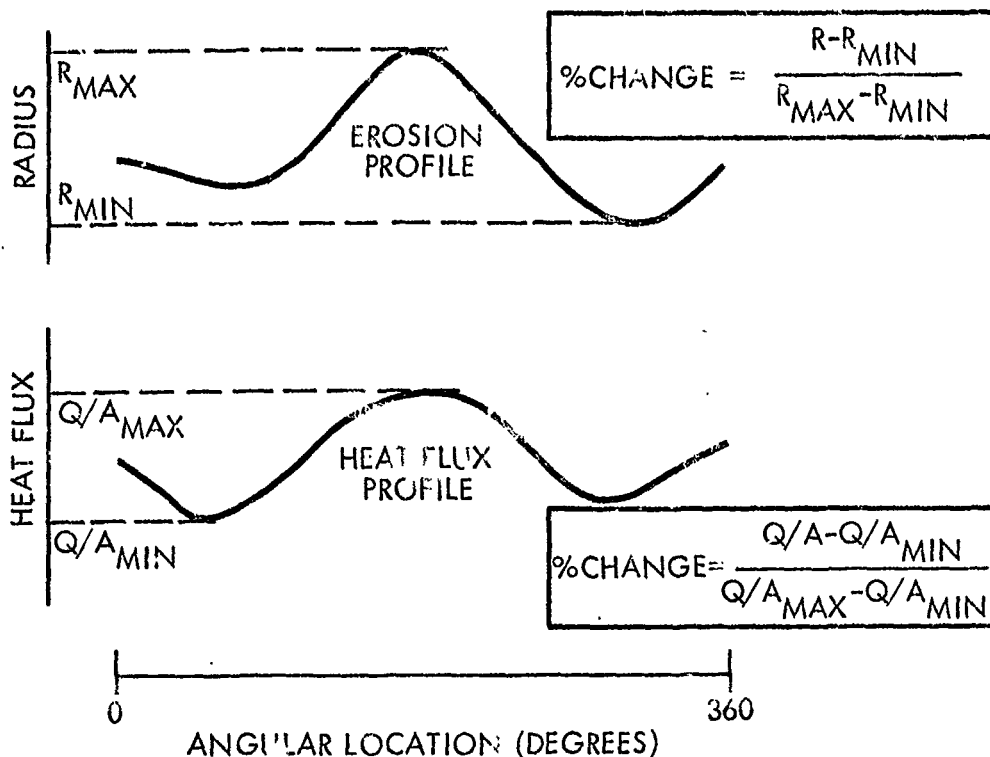
The Qual B Baseline experimental heat transfer studies were conducted using HEA 023. The erosion data were obtained from the ablative erosion baseline tests and streak chamber tests conducted at 25% thrust and at FTP.

- A significant erosion/thermal environment correlation was established using the 25% thrust streak and ablative chamber throat erosion profiles and the heat flux data acquired from the 25% thrust heat sink test. These data show that the percent change in heat flux at the throat produces approximately the same percent change in erosion radius.
- FTP, 65%, and 50% thrust levels show generally that the ablative erosion radius increases as heat flux increases. However, no clear-cut correlations were evident.
- There was no satisfactory correlation between heat fluxes in the throat region of the aluminum water-cooled chamber and erosion depths measured at the throat in the ablative and streak chambers.
- Peak heat fluxes in the aluminum chamber throat region were recorded at angular locations between 110 and 120°. This is in good accord with heat sink and erosion data.

Thermal Environment/Erosion Correlation Results

The results of the erosion/heat flux profile studies are presented in Figures 3-79 through 3-84. The percent changes shown in the results are the changes relative to the minimum values of erosion radius or heat flux measured circumferentially around the throat.

The percent changes in erosion radius and heat flux shown in Figures 3-82, 3-83, and 3-84 were calculated from data shown in Figures 3-79, 3-80, and 3-81. The calculation procedure was as follows:



The results obtained and the analysis performed for each thrust level are presented as follows:

- 25% Thrust

Figure 3-79 shows the data utilized to accomplish the erosion/thermal environment correlation. The data was comprised of (1) the 25% streak (VA1-665) throat erosion profile, (2) the duty cycle tested ablative chamber (VA1-666 thru 671) throat erosion profile, and (3) the 25% H/S test data from VA1-692. The 25% streak had been tested specifically to aid in the erosion profile correlation quite early in the testing program. Also, since the final portion of the duty cycle imposed on the ablative chamber was at 25% thrust (see Figure 3-85) a wall erosion profile indicative of 25% was probably created, as opposed to the profile being highly dependent upon thrust level excursions preceding the final portion of the duty cycle.

Figure 3-82 shows the correlation at 25% thrust which indicates a one-to-one relationship between erosion radius changes at the throat to the indicated heat flux changes for both streak and ablative chamber throat erosion profiles.

- FTP

Figure 3-80 shows the data utilized at FTP to compare erosion trends with heat flux. The data consisted of (1) the FTP streak (VA1-659) throat erosion profile (2) the duty cycle tested ablative chamber (VA1-666 thru 671) throat erosion profile, and (3) the FTP H/S test data from VA1-693. The FTP streak had been tested to provide erosion characteristics to aid in possible correlations with heat transfer data at the throat.

Figure 3-83 shows the percent variations of heat flux and erosion radius at FTP for both the streak and ablative chamber throats. As indicated there is no clear-cut correlation. Since the ablative erosion was a post duty cycle profile measurement, it could not be anticipated that a correlation would exist. The FTP portion of the duty cycle shows only a 4% throat area change; whereas, the throat area had increased 24% by the end of the duty cycle.

- 50% and 65% Thrust

Figure 3-81 shows the duty cycle tested ablative throat erosion profile, the 50% thrust H/S chamber test data, and the 65% thrust H/S chamber test data.

Figure 3-84 shows the percent variation of heat flux and erosion radius for the ablative throat for both 50% and 65% thrust. There is some heat flux/erosion correlation evidence shown but it cannot be construed to mean that the correlations are valid. Since both thrust levels occur in the middle of the duty cycle it was not anticipated that the erosion profile would be indicative of 50% and 65% heat flux profiles (unless of course, the heat flux profiles at each duty cycle thrust level matched exactly).

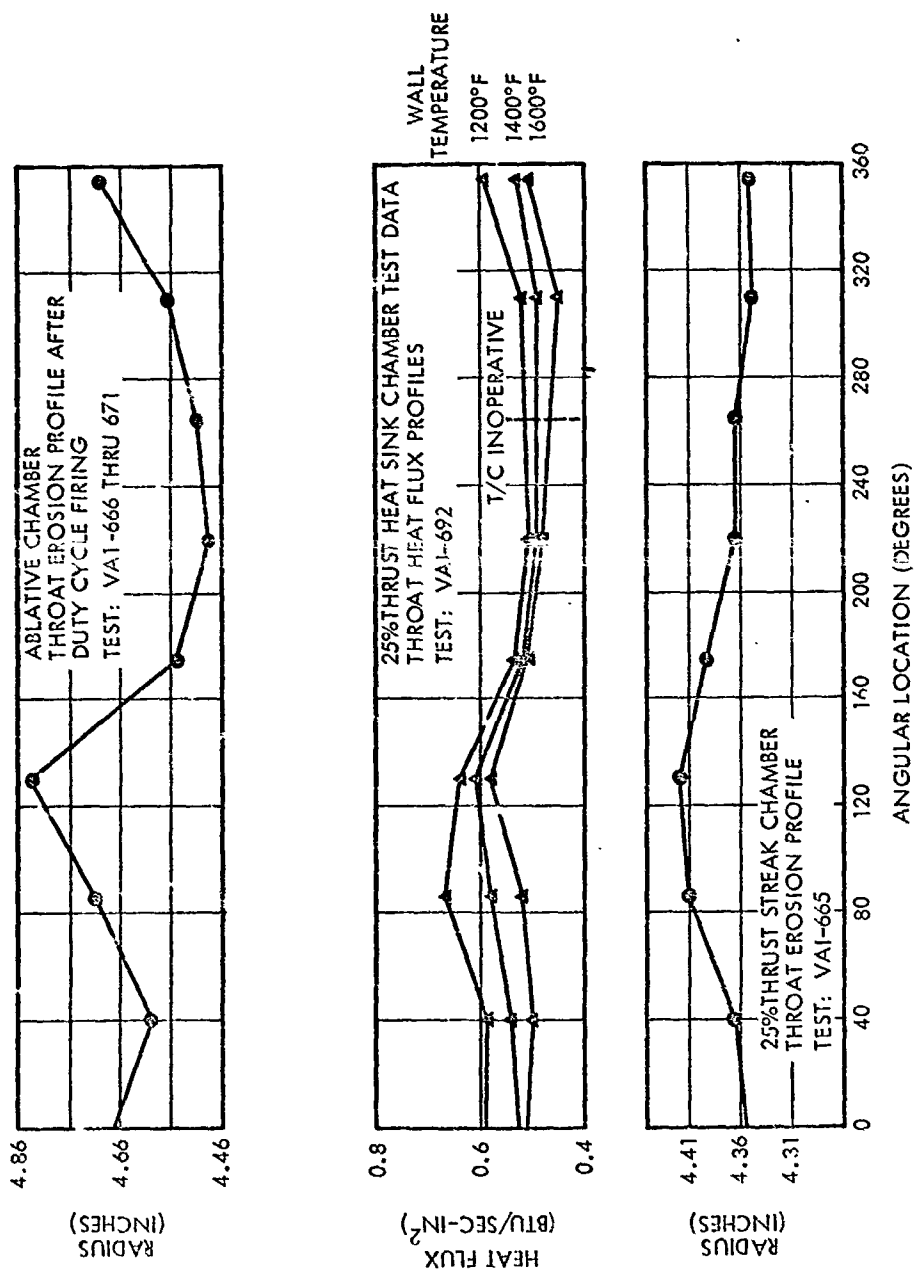


FIGURE 3-79 THROAT EROSION AND HEAT FLUX PROFILES, 25% THRUST

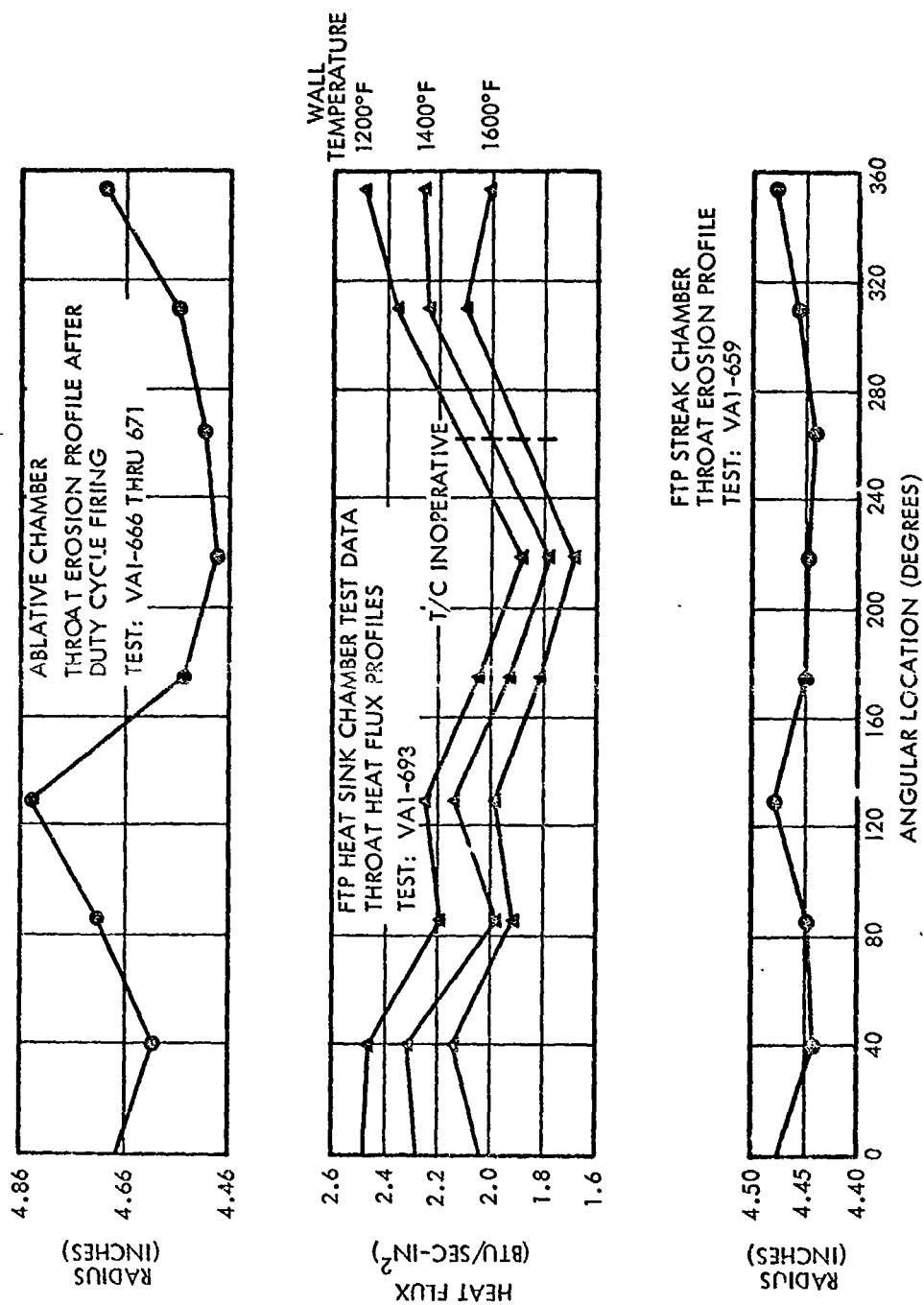


FIGURE 3-80 THROAT EROSION AND HEAT FLUX PROFILES, FTP

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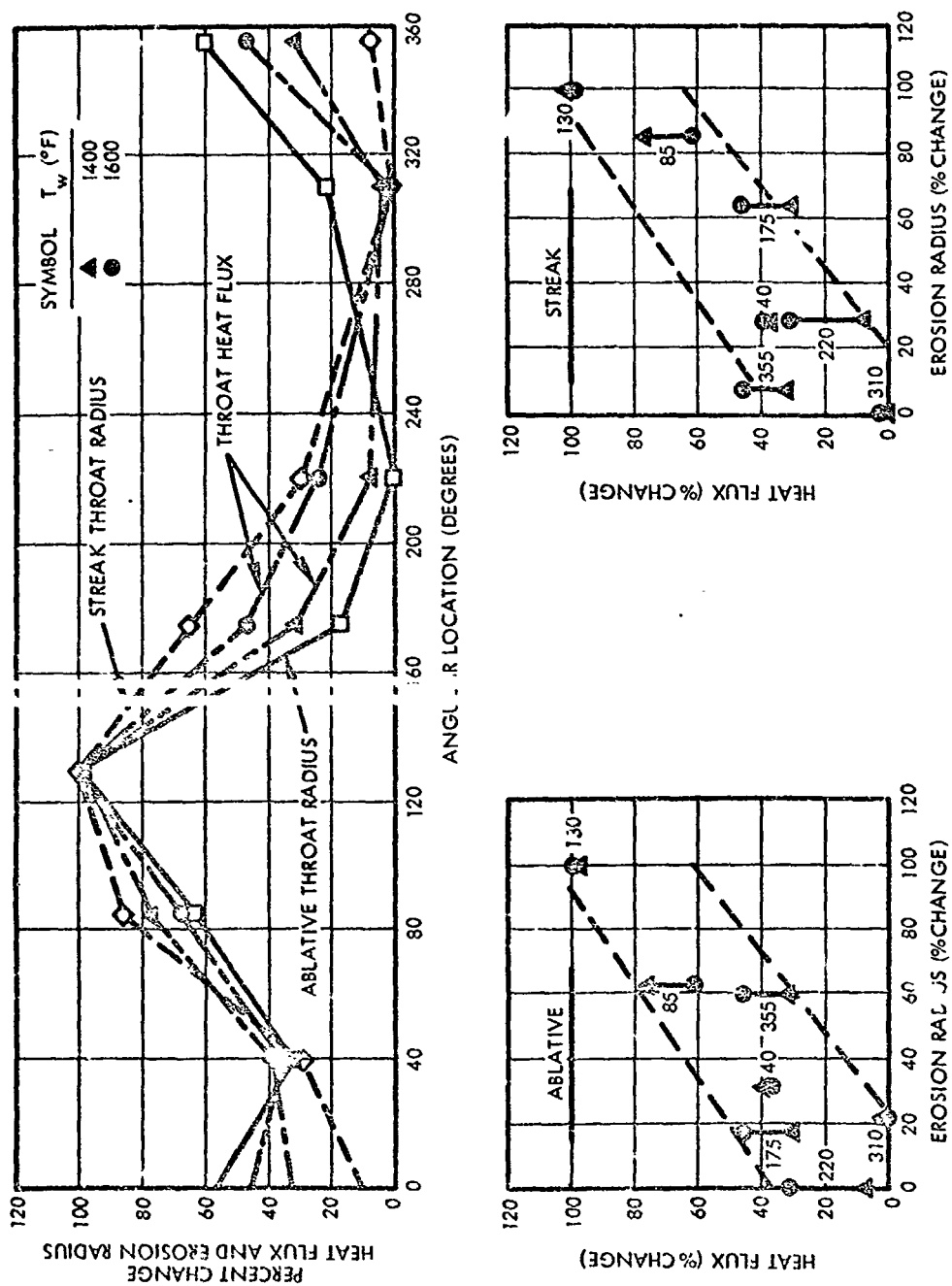


FIGURE 3-82 THROAT EROSION/H.T. FLUX CORRELATION, 25% THRUST

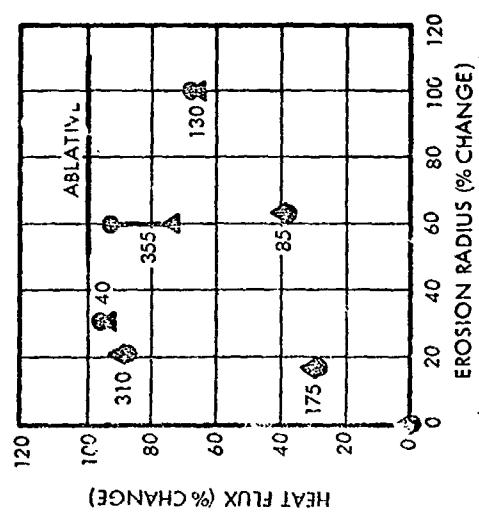
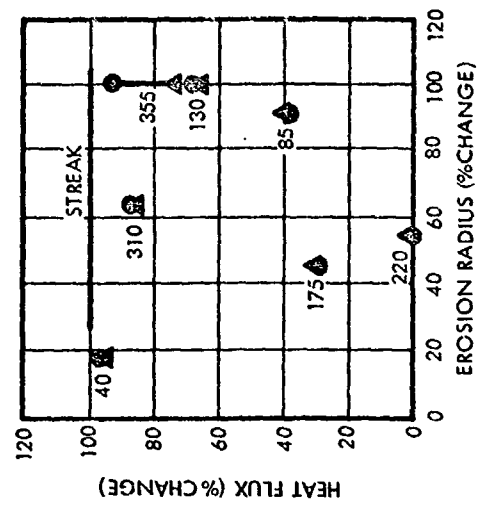
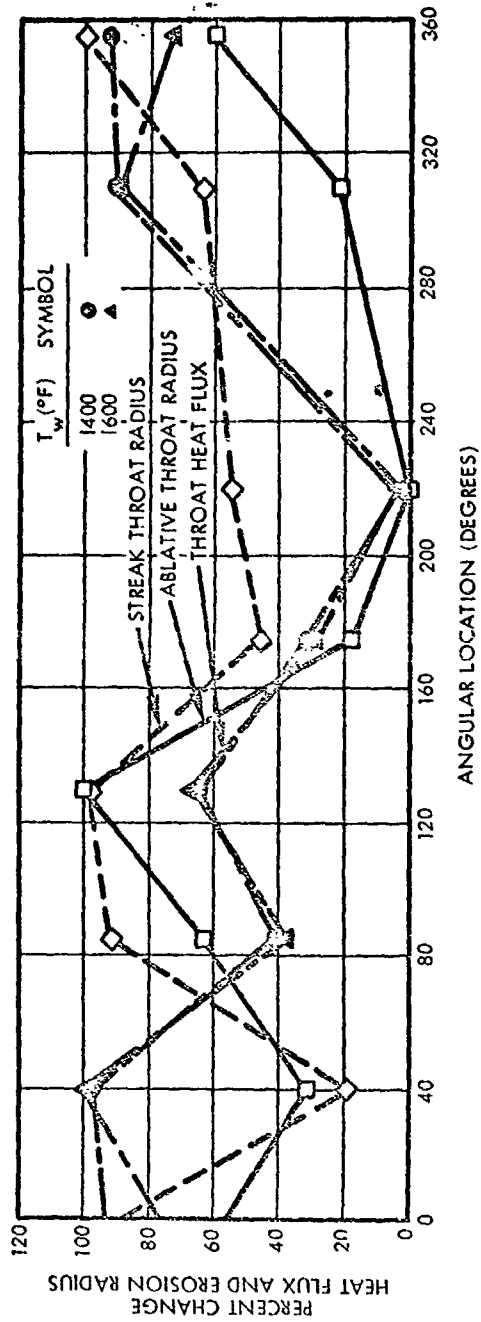


FIGURE 3-83 THROAT EROSION/HEAT FLUX CORRELATION, FTP

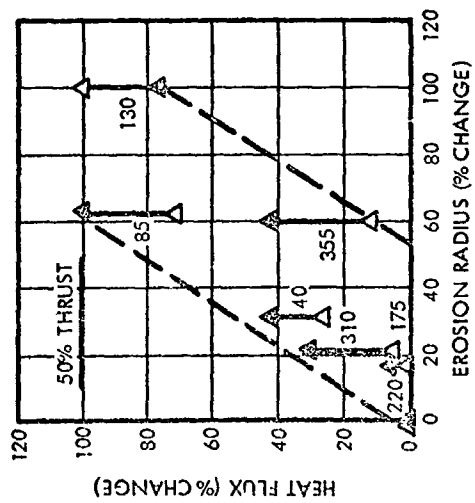
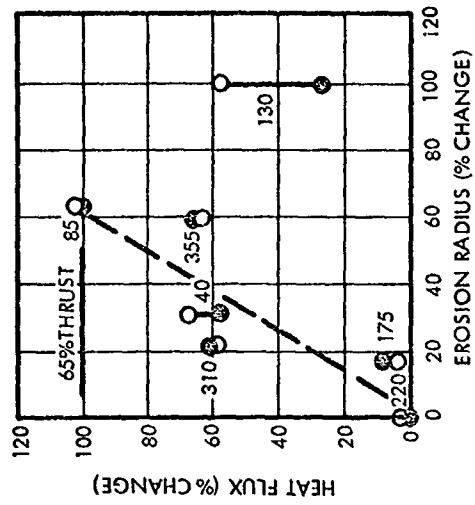
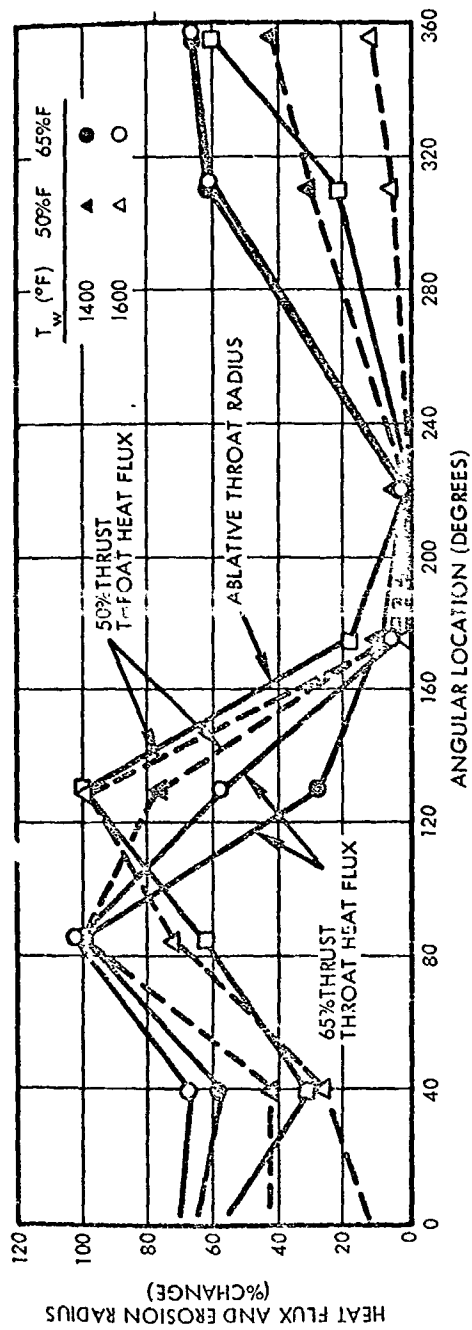


FIGURE 3-84 THROAT EROSION/HEAT FLUX CORRELATION, 50% AND 65% THRUST

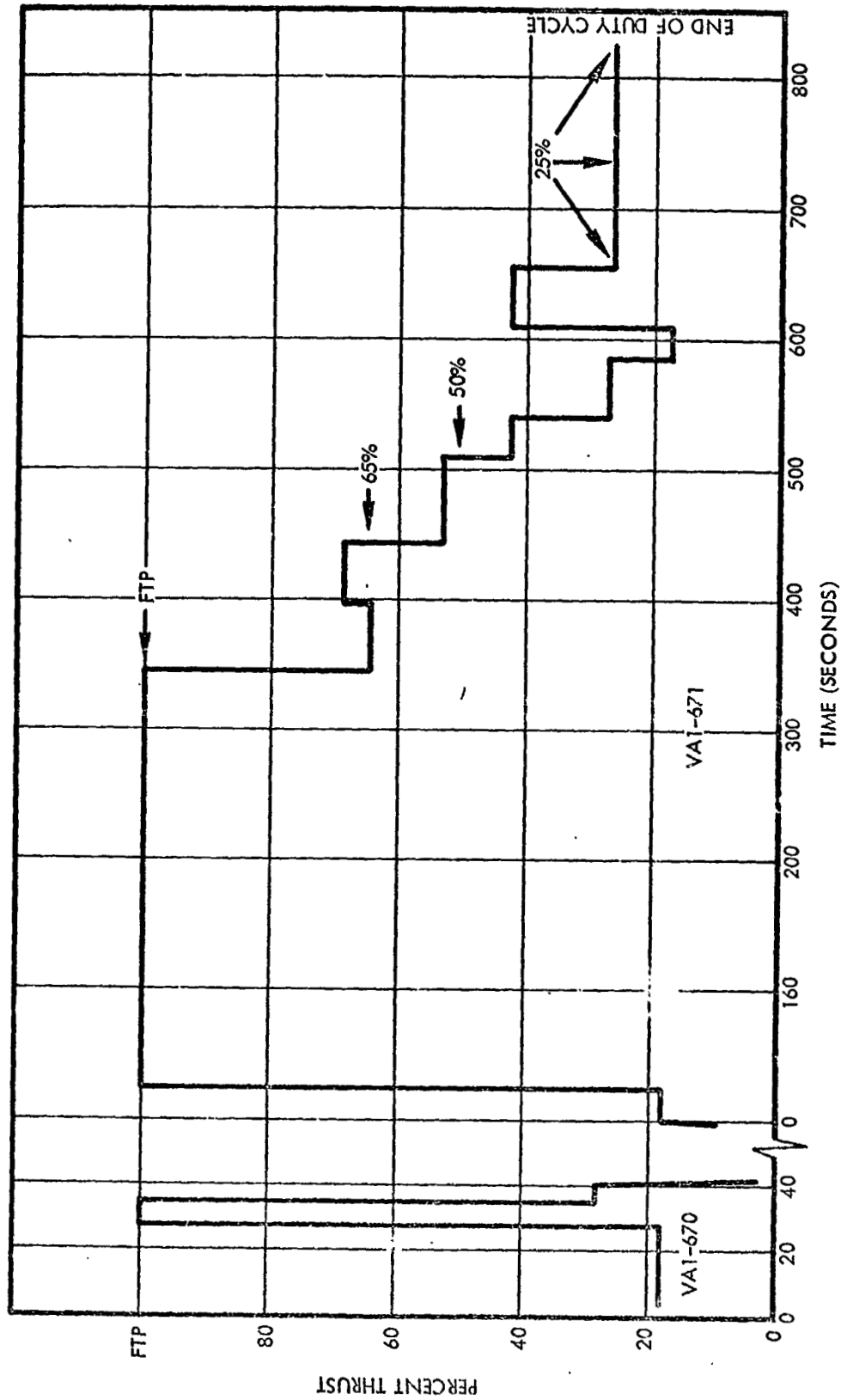


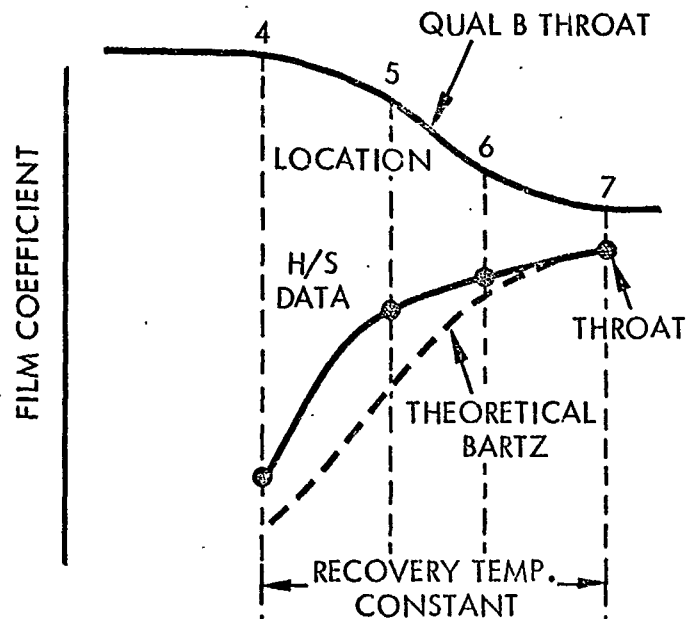
FIGURE 3-85 ABLATIVE CHAMBER DUTY CYCLE

Aluminum Rig Heat Flux/Erosion Comparisons

An attempt was made to correlate the aluminum rig heat flux data with erosion data obtained from the ablative erosion baseline tests and streak chamber tests conducted at 25% thrust and at FTP.

No satisfactory correlation could be found; although the location of maximum heat flux at angular location 110 - 120° corresponds to the location of maximum erosion. The lack of correlation is the result of a number of factors, namely:

- The "throat" region investigated in the aluminum water-cooled chamber consists of more than just the small region in the immediate vicinity of the throat plane and extends upstream into the chamber. The region immediately upstream of the throat plane has exhibited heat flux variations (heat sink data of Figure 3-86) which indicate high turbulence or flow impingement effects as shown below.



The flow pattern in this upstream region is not as clearly defined as at the throat itself where erosion measurements are usually made. Since the throat heat flux is actually an average over a region where rapid heat flux changes probably occur axially and circumferentially it is not surprising that no satisfactory erosion correlation could be found using the aluminum water-cooled throat data.

NOTE: EACH POINT SHOWN
BELOW REPRESENTS THE AVERAGE OF
EIGHT (8) CIRCUMFERENTIAL
DATA POINTS

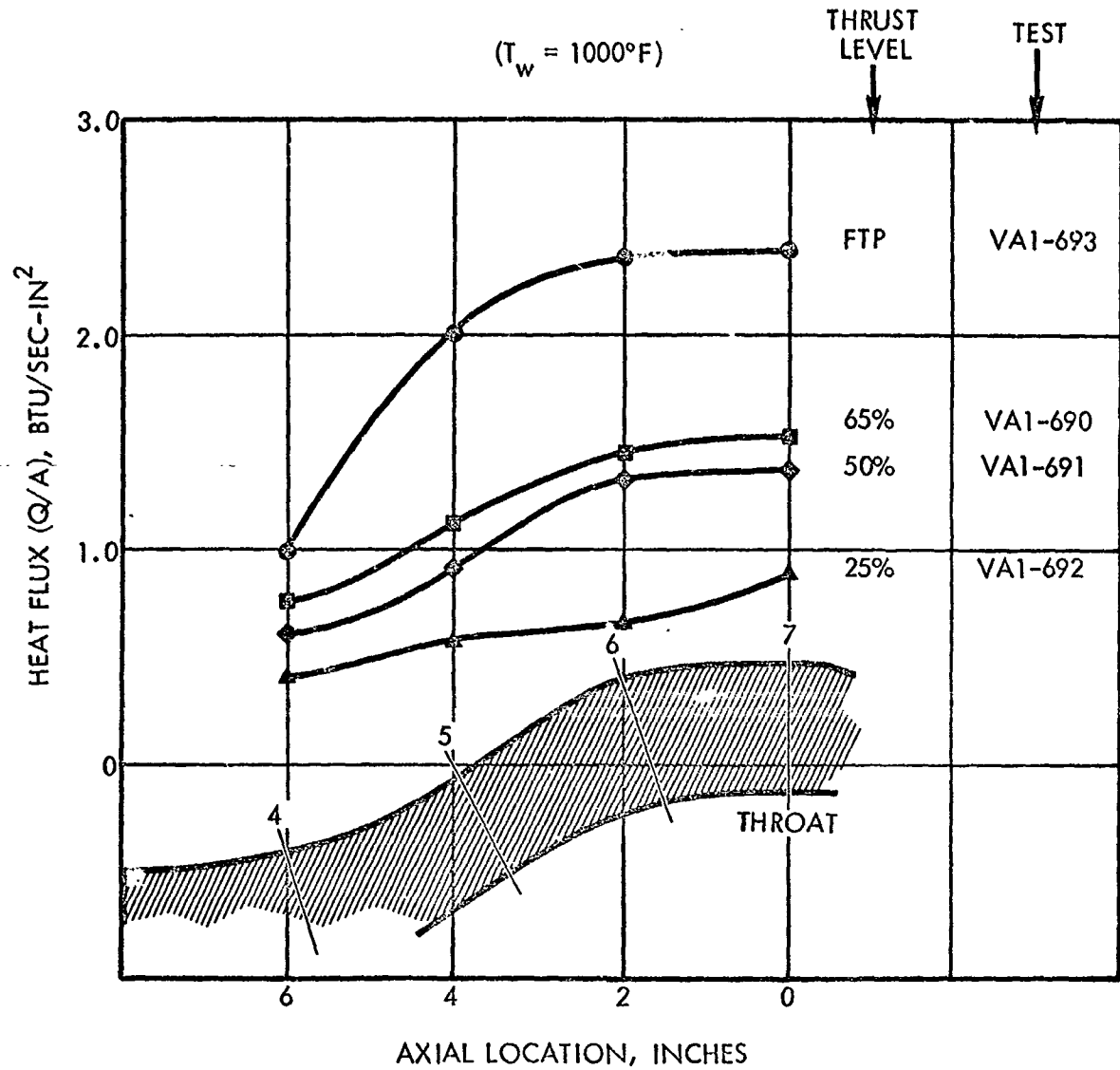


FIGURE 3-86 HEAT SINK CHAMBER THROAT REGION HEAT FLUX PROFILES

- Due to the low wall temperature, the chemical environment in the immediate vicinity of the wall may be considerably different than that under which ablation actually occurs. Since ablation is both a thermal and chemical phenomenon this factor can be quite significant.

The throat heat flux/erosion profile study approach was used to obtain information and correlations which might show the predictability of using heat flux as a measurement of anticipated throat erosion. The validity of using heat flux changes to accurately predict erosion changes appears to have some merit. However, the extent to which such predictions might be carried is yet to be determined.

3.8.2 Improved Flow Distribution Test Series

This section presents the correlation between throat heat flux and erosion test data for Contour 'A' and Contour 'C' chambers at FTP using the XX2 injector. The heat flux information was derived from data obtained during the improved flow distribution test series. The erosion data was obtained from 60 second streak chamber tests conducted at FTP as part of the same test series.

Summary

Correlation of heat flux data with streak erosion information has yielded the following:

- A significant heat flux/erosion correlation has been established for the Contour 'A' chamber at FTP. The data shows that the percent change in heat flux at the throat produces approximately the same percent change in erosion radius.
- No satisfactory heat flux/erosion correlation could be established for the Contour 'C' chamber at FTP. However, erosion of the chamber turbulence ring during each firing probably alters the flow distribution from test to test thus precluding exact repeatability in circumferential heat flux from test to test.

Results

Figure 3-87 presents data obtained from tests on chamber Contour 'A' with the XX2 injector. Throat radius subsequent to streak chamber firing (VA2-643) and heat flux from heat sink firings (VA2-648) are plotted as a function of circumferential location. It can be seen that these curves have similar shapes indicating that a correlation may be practical.

Correlation of the two sets of data was performed by the technique described in Section 3.8.1 and the results of this correlation are shown in Figures 3-88 and 3-89. Figure 3-88 expresses percent changes in heat flux and erosion radius relative to the minimum value of each. Figure 3-89 plots percent erosion radius change against percent heat flux change.

These results indicate a very good correlation between the two sets of data.

Figure 3-90 presents data obtained from tests on chamber Contour 'C' with the XX2 injector. Throat radius subsequent to streak chamber firing (VA2-653) and heat flux from heat sink firings (VA2-654) are plotted as a function of circumferential location. Examination of the plots reveals the following:

- The two curves do not have the same shape indicating that a satisfactory correlation may not be possible.
- The difference between maximum and minimum values for both heat flux and erosion radius are significantly smaller than for Contour 'A'. This is probably the effect of the large turbulence ring close to the head end of Contour 'C' which, by turbulating the flow, results in a more uniform circumferential flow distribution.

Figures 3-91 and 3-92 present the results of the attempted correlation of the two sets of data for Contour 'C' and confirm the curves of Figure 3-90. No satisfactory correlation could be found. The primary cause of the lack of correlation probably lies in the presence of the large turbulence ring in Contour 'C'. In addition to smoothing out the flow,

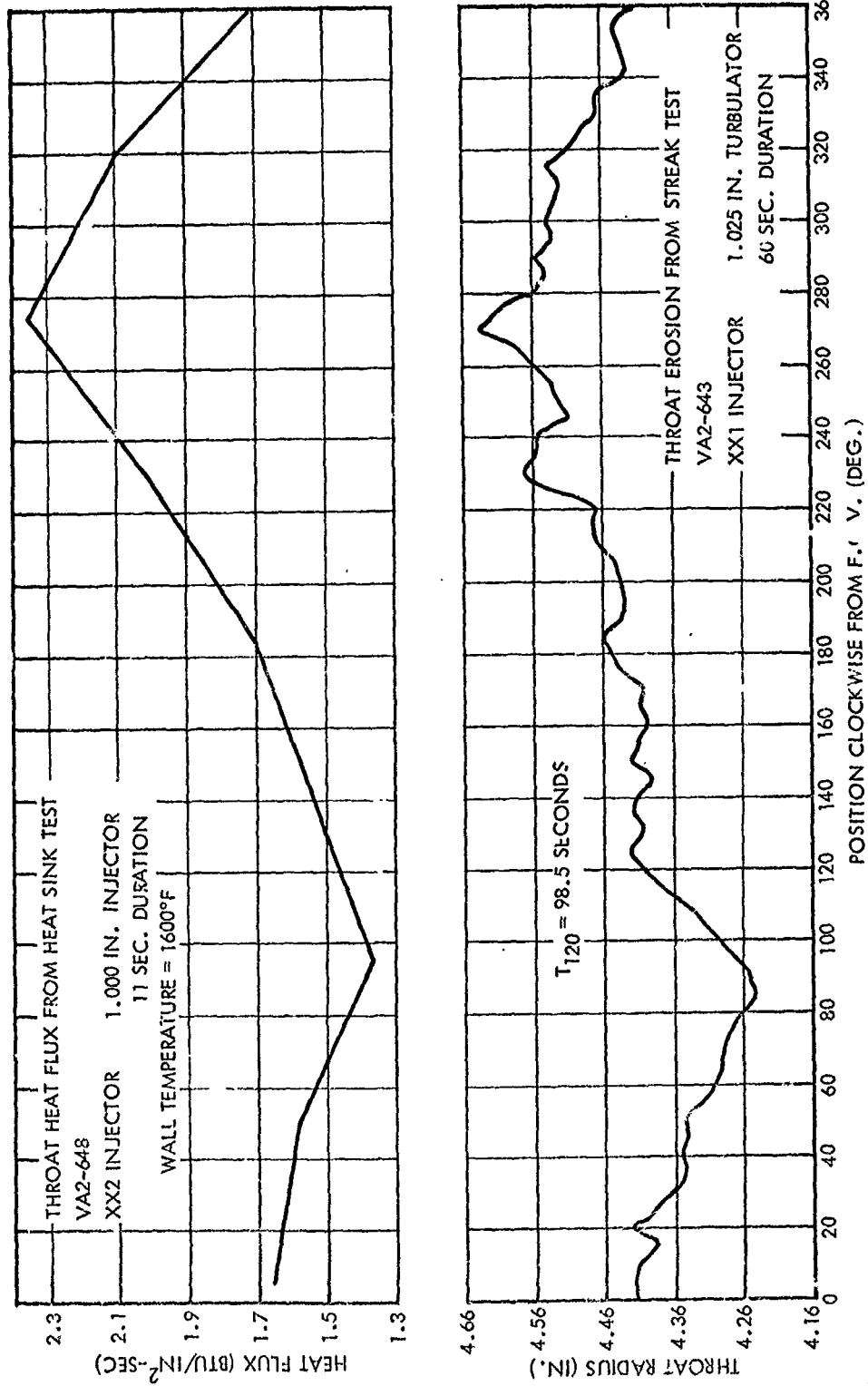


FIGURE 3-87 THROAT EROSION AND HEAT FLUX PROFILES, CONTOUR "A", FTP

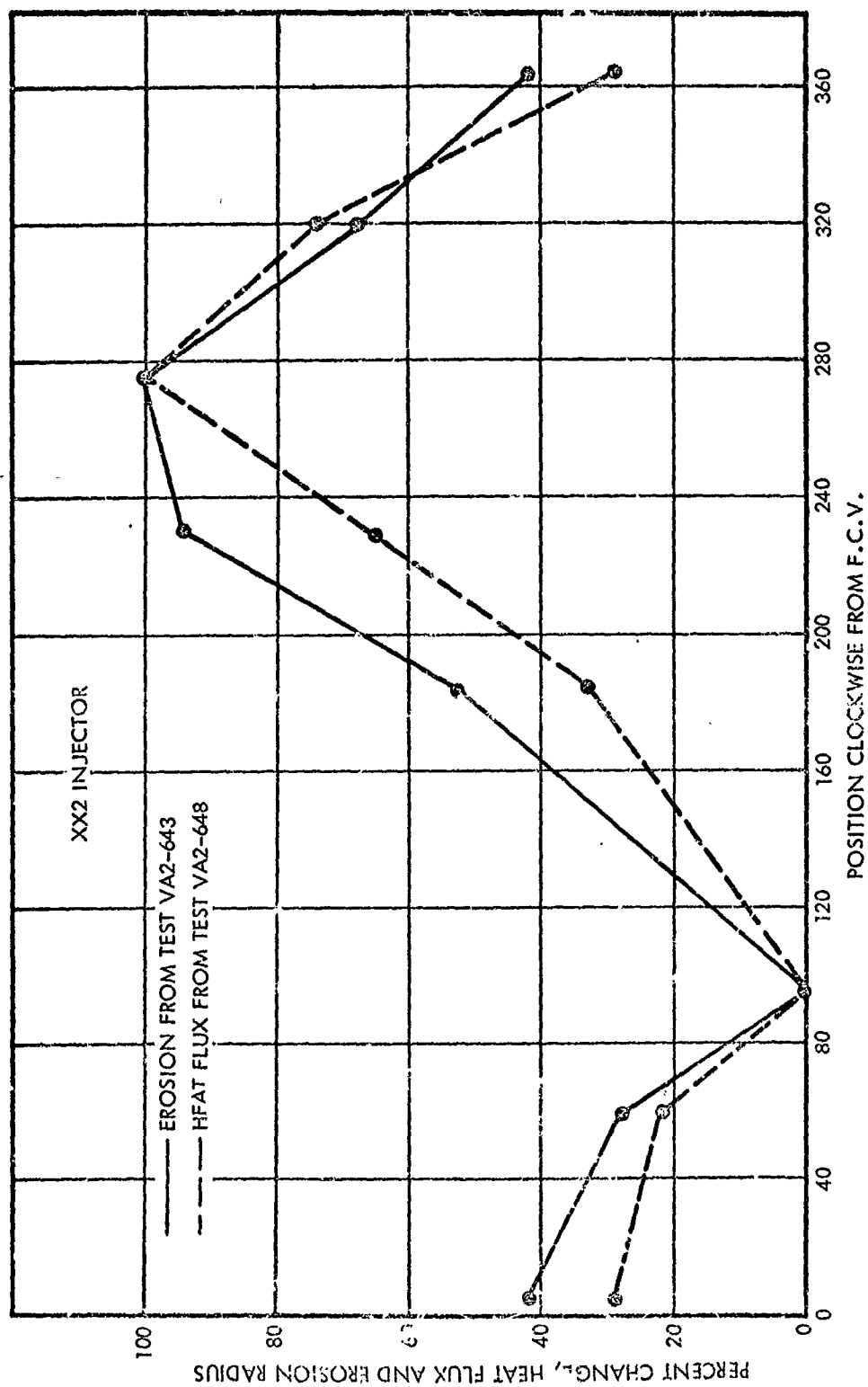


FIGURE 3-88 THROAT EROSION AND HEAT FLUX CHANGE PROFILES, CONTOUR "A", FTP

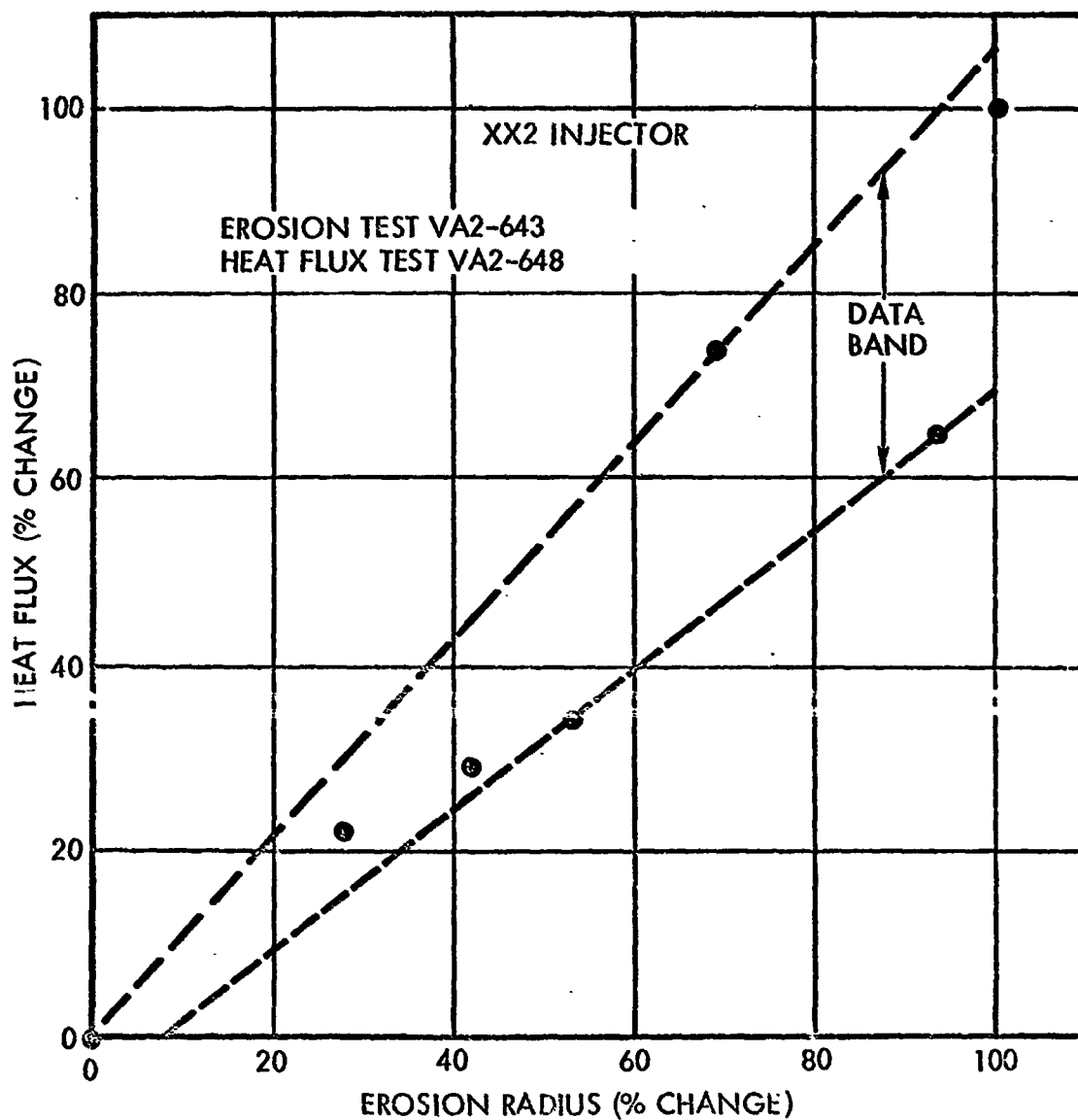


FIGURE 3-89 THROAT EROSION/HEAT FLUX CORRELATION, CONTOUR "A", FTP

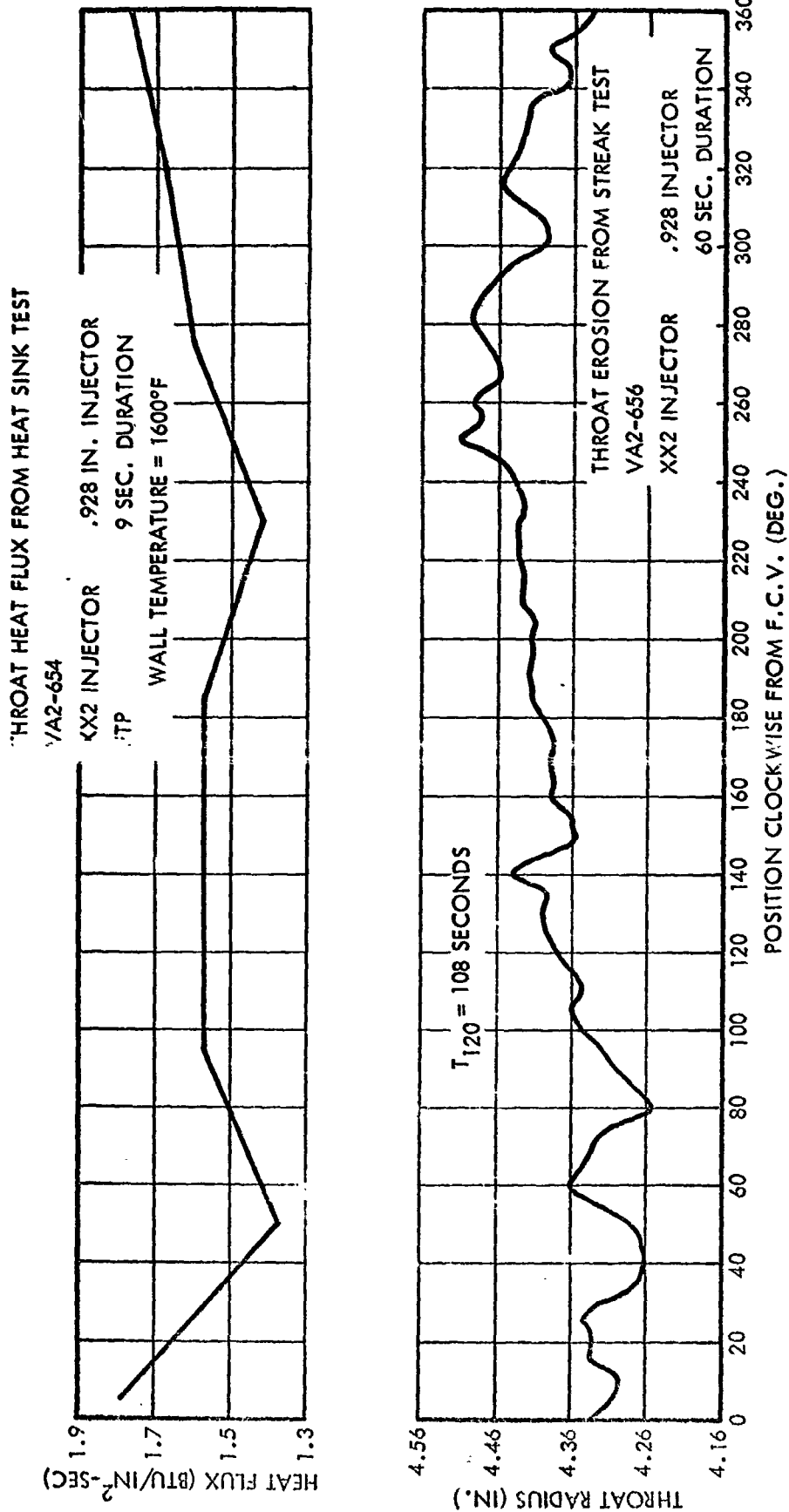


FIGURE 3-90 THROAT EROSION AND HEAT FLUX PROFILES, CONTOUR "C", FTP

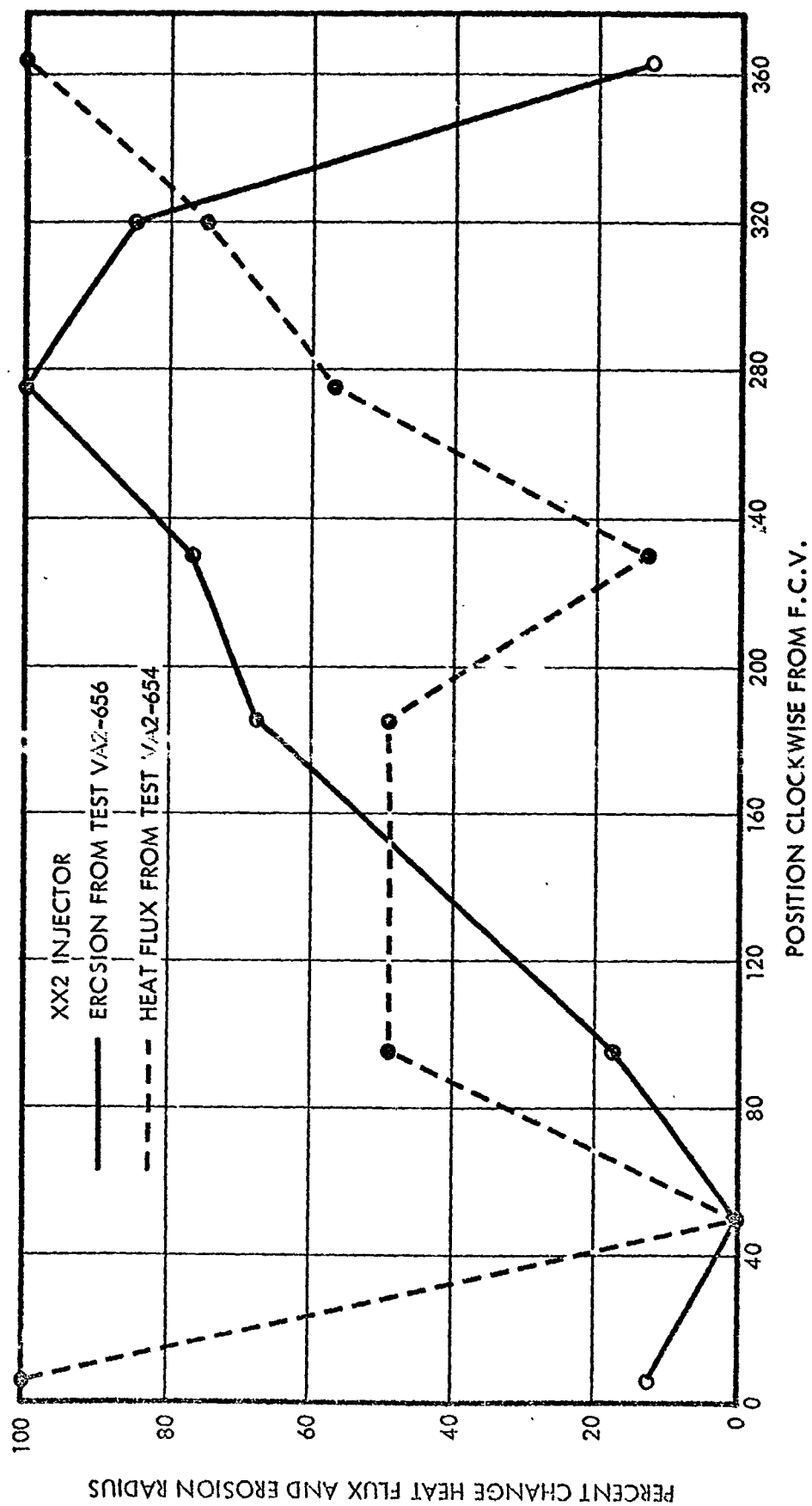


FIGURE 3-91 THROAT EROSION AND HEAT FLUX CHANGE PROFILES, CONTOUR "C", FTP

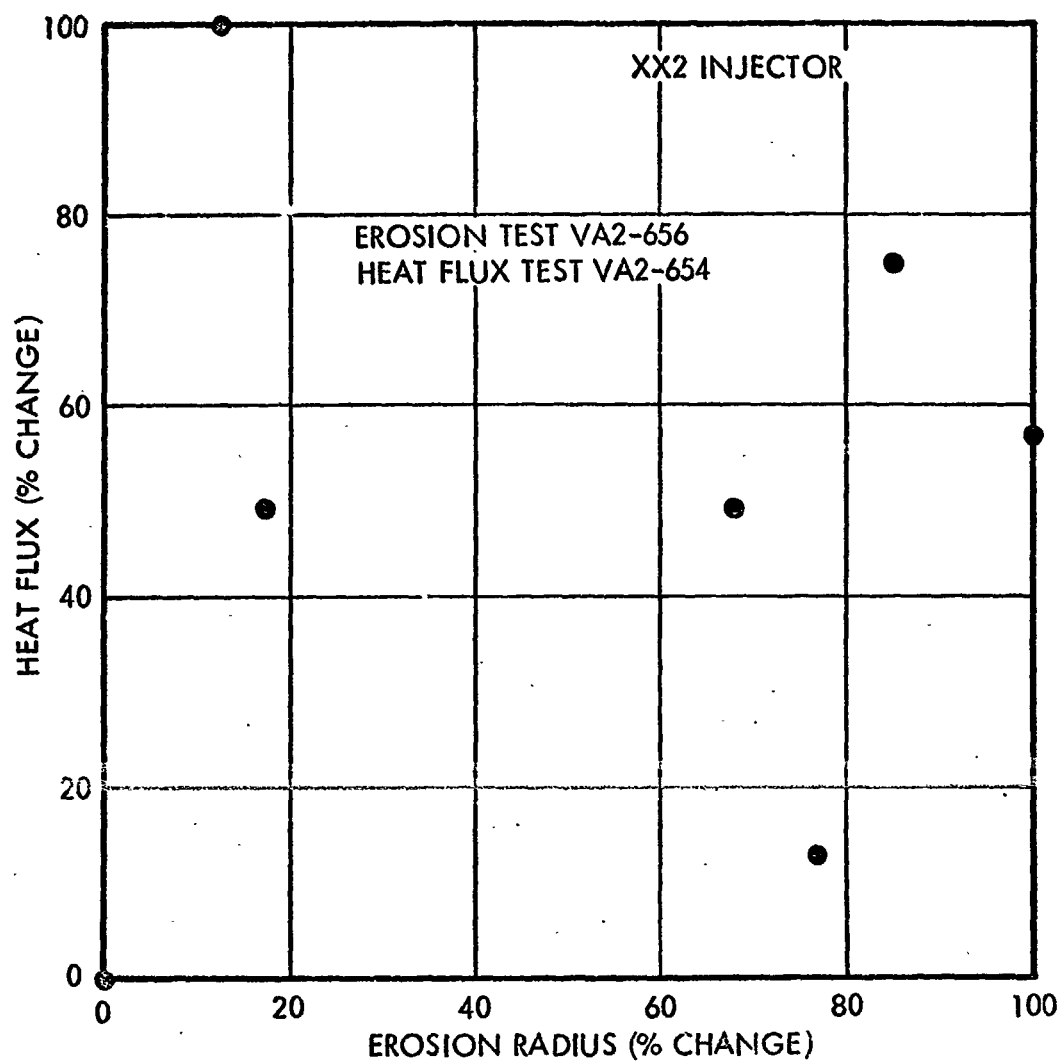


FIGURE 3-92 THROAT HEAT FLUX/EROSION CORRELATION, CONTOUR "C", FTP

sectioning of the chamber after the tests revealed significant erosion of the ring during the tests. Since the erosion was not uniform this indicates that the flow pattern along the walls may vary from test to test with little possibility of attaining exact repeatability.

3.9 "T₁₂₀" EROSION INDEX

Erosion characteristics of LMDE head end assemblies are established by reactive testing with a fiberglass streak nozzle at a single throttle point, usually FTP. The index of erosion is the "T₁₂₀ time", the run time equivalent to 20 percent throat area increase for the streak sample. This index is used throughout this report.

T₁₂₀ values are calculated between any two selected test data points during the run as follows:

$$T_{120} = \frac{(0.20) (t_2 - t_1)}{c^*_1/c^*_2 - 1.0}$$

Where: t = time, seconds

c* = chamber characteristic exhaust velocity, ft/sec

Subscript 1 = condition at first time slice

Subscript 2 = condition at second time slice

The "T₁₂₀ time" values for the FTP streak erosion tests are calculated from the 15.0-second and the 58.0-second data points, or the last steady-state data point in the run. The 15.0-second data point was selected because the transient heat effects have stabilized. For the 25% thrust streak erosion tests, the T₁₂₀ values are calculated from the 58.0-second and 98.0-second data points.

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3.10 OXIDIZER TURBULATOR EFFECTS

The effect of oxidizer turbulator size was evaluated during three test series in the course of the program.

HEA 023 was initially assembled with a 1.025-inch-diameter turbulator which was subsequently replaced with a 1.000-inch-diameter turbulator after the first fiberglass streak test (VA1-657) produced a low "T₁₂₀ time" of 69 seconds. One of the objectives of the test series was to adjust turbulator size such that the fiberglass streak erosion "T₁₂₀ time" fell within the range of 75 to 85 seconds. The 1.000-inch-diameter turbulator produced an acceptable "T₁₂₀ time" of 79 seconds (VA1-659) with a near nominal mixture ratio of 1.598.

The characteristic velocity (c*) with a water-cooled chamber was the same with the 1.000-inch-diameter turbulator (VA1-656) as with the 1.025 inch diameter turbulator (VA1-661). The 10 second increase in "T₁₂₀ time" is equivalent to a reduction in the predicted FTP throat erosion for 375 seconds from 16 percent to 8 percent with a Qual B chamber assembly.

During the injector flow distribution test series with injector XX2, four different sized turbulators were used with a Contour A chamber assembly containing a heat sink throat. The chamber heat flux as a function of turbulator diameter is discussed in Section 3.5.5.4. The uncorrected FTP c* varies from 5384 ft/sec for the 1.025-inch-diameter turbulator to 5350 ft/sec for the 0.925-inch-diameter turbulator, as shown in Table 3-10.

TABLE 3-10
EFFECT OF TURBULATOR DIAMETER ON PERFORMANCE

TEST NO.	TURBULATOR DIA. (INCH)	c* FT/SEC	M.R.
VA2-644	1.025	5384	1.600
VA2-648	1.000	5380	1.599
VA2-652	0.960	5367	1.598
VA2-649	0.925	5350	1.602

During this same flow distribution test series, two different turbulator diameters were used with a Contour C chamber assembly containing a heat sink throat. The uncorrected c^* varied from 5502 ft/sec (VA2-654) with a 0.925-inch-diameter turbulator to 5550 ft/sec (VA2-655) with a 1.000-inch-diameter turbulator.

Figure 3-93 contains curves of oxidizer turbulator area versus c^* for both the Contour A and Contour C chamber assemblies based on the data from the tests specified above.

Prior to conducting a duty cycle on a Contour C quartz chamber assembly in the HATS facility (Section 5.4), HEA 034 was tested on VETS with a water cooled chamber using two different sized turbulators. Two different sized turbulators were also used during the checkout tests on HATS prior to the duty cycle. During the VETS testing with HEA 034, the Qual B contour water-cooled chamber c^* varied from 5440 ft/sec (VA2-690) with a 1.000-inch-diameter turbulator to 5405 ft/sec with a 0.925-inch-diameter turbulator, as shown in Figure 3-93. When the turbulator diameter was changed from 1.000 inch to 0.925 inch during HATS testing, the I_{sp} at FIP dropped from 304.2 seconds (HATS-417) to 301.2 seconds (HATS-418). This 3 second drop in I_{sp} , or one percent performance decrease, agrees very well with the 48 ft/sec drop in c^* (one percent) experienced with the Contour C heat sink chamber when going from a 1.000-inch to 0.925-inch-diameter turbulator as shown in Figure 3-93. Figure 3-93 also shows that performance decreases at a more rapid rate for a Contour C chamber than for a Contour A or Qual B chamber as the turbulator diameter is reduced. As turbulator diameter increases, there appears to be a limiting size beyond which there is little, if any, effect on engine c^* performance.

3.11 THROAT PRESSURE PROFILE

During the flow distribution test series (Section 3.5.5.4), one heat sink throat was instrumented with pressure taps at twelve axial locations, Figure 3-94, with the objective of determining whether performance differences as indicated by head-end chamber pressure (P_{ch}) were real. Contour A and Contour C were compared in the test series. Contour C had an apparent increased performance as measured by P_{ch} .

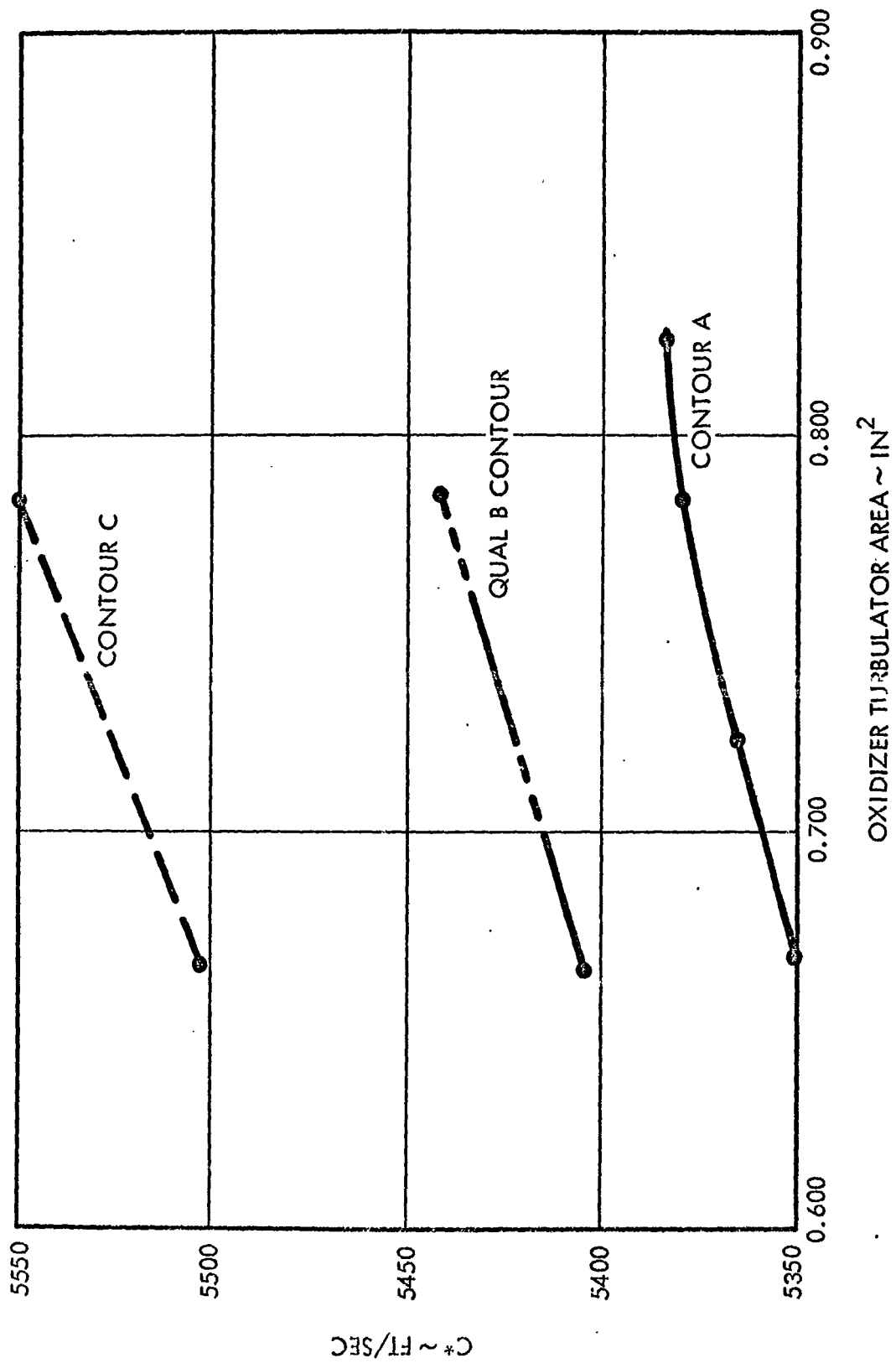


FIGURE 3-93. PERFORMANCE EFFECT OF TURBULATOR SIZE

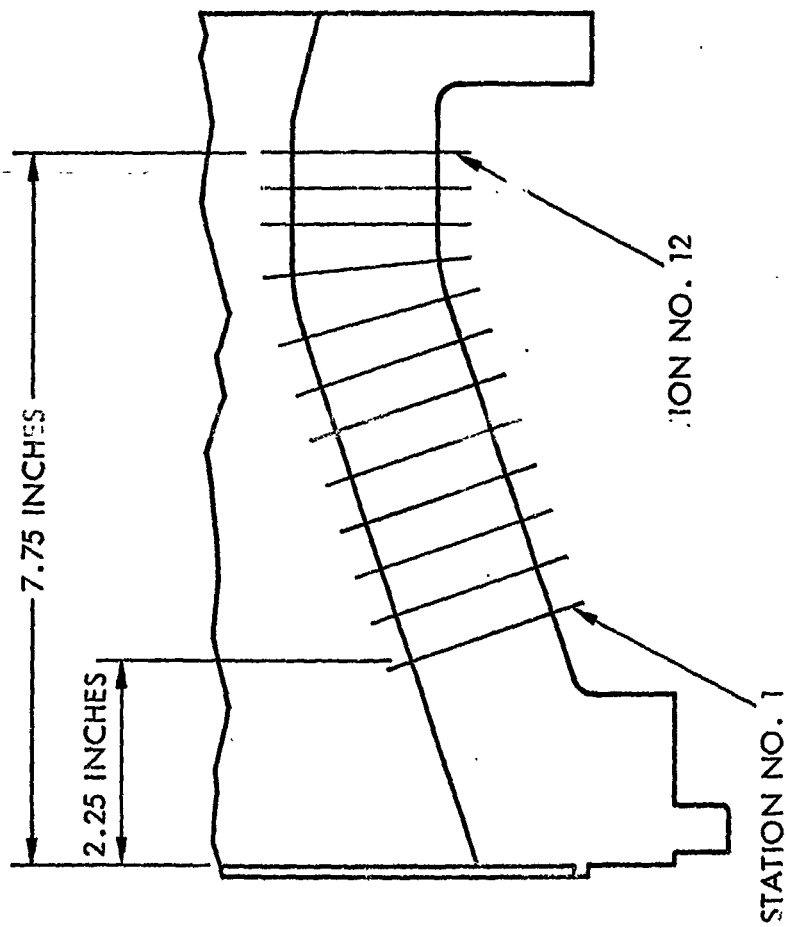


FIGURE 3-94. HEAT SINK THROAT - STATIC PRESSURE TAP LOCATION

Static pressures for 3 FTP tests are plotted in Figure 3-95 from data normalized to a total propellant flow rate of 32.4 lb/sec. The data are tabulated in Table 3-11.

It was concluded that the static pressure method was too insensitive to detect performance variations of less than one percent. For instance, pressures at Stations 10, 11 and 12 show no performance difference for Contour C with the 0.925 and 1.000-inch-diameter turbulator, but at stations farther upstream the 1.000-inch-diameter turbulator appears to have an advantage. As a result, the static pressure performance measurement approach was terminated.

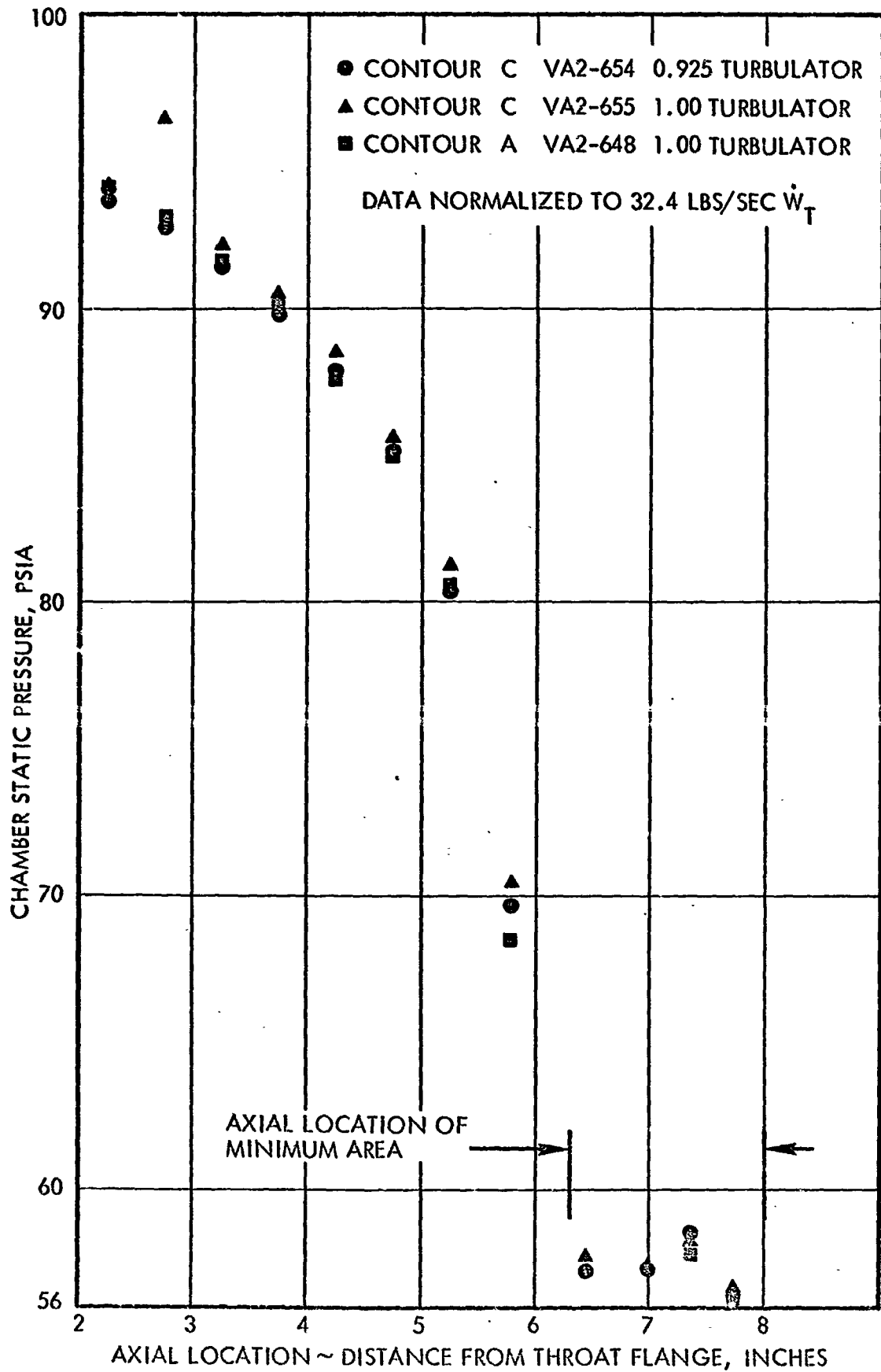


FIGURE 3-95. CHAMBER STATIC PRESSURE VS AXIAL LOCATION

TABLE 3-11
CHAMBER STATIC PRESSURE (PSIA)

LOCATION	VA2-654 CONTOUR C 0.925 IN. DIAM. TURB.	VA2-655 CONTOUR C 1.000 IN. DIAM. TURB.	VA2-648 CONTOUR A 1.000 IN. DIAM. TURB.
P _{ch} Head End	106.6	106.4	103.1
Throat Station*			
1	93.7	94.2	94.3
2	92.8	96.5	93.1
3	91.5	92.2	91.6
4	89.9	90.5	90
5	87.8	88.5	87.6
6	85.1	85.5	85.1
7	80.4	81.2	80.6
8	69.7	70.4	68.5
9	57.2	57.6	56.1
10	57.3	57.3	56.4
11	58.4	58.3	57.9
12	56.3	56.5	56.2

*REF: FIGURE 3-94

NOTE: DATA NORMALIZED TO 32.4 LBS/SEC TOTAL PROPELLANT FLOW

4. HARD THROAT THERMAL ANALYSIS

4.1 ANALYSIS OF CANDIDATE MATERIALS

A preliminary thermal analysis of the four candidate hard throat insert materials was completed. The analysis was intended to be used only for screening purposes to aid in the selection of a candidate insert material. The four selected insert materials were Ta-W, JTA graphite, W-Re/ZrO₂ composite and Pyro-Carb. The thermal analysis of the insert materials was conducted, on a parametric basis, to evaluate the effect of variations of material thickness on insert thermal gradients, in-wall temperatures, back-wall, and surface operating temperatures. The LMDE (Qual B) mission duty cycle was used for the imposed thermal environment.

To evaluate the relative thermal merits of the four materials the following design criteria were used; (1) maintain the backwall temperature during the firing at less than 800°F to minimize the thermal stresses in the titanium outer shell, (2) maintain the interface temperature between the insert material and the MX-2600 less than 3000°F to prevent high temperature ablation of the MX-2600, and (3) minimize the temperature gradient across the insert to prevent severe thermal shock loading. Another thermal design constraint limits interface temperature at less than 700°F during the 33 second initial burn and the following one hour coast. This will prevent charring of the MX-2600 prior to the duty cycle final burn. If charring occurred the resultant outgassing could result in loosening of the insert with possible detrimental effects during the final burn.

4.1.1 Interface Temperature Study

A thermal analysis was performed to determine the interface temperatures between the four hard throat insert materials and the MX2600 ablative during the initial firing and the subsequent soakback. Parametric studies were performed to investigate the effect of insert thickness as well as inserting various thicknesses of pyrolytic graphite between the insert and the ablative. The LMDE (Qual B) mission duty cycle of Table 4-1 was used for the imposed thermal environment.

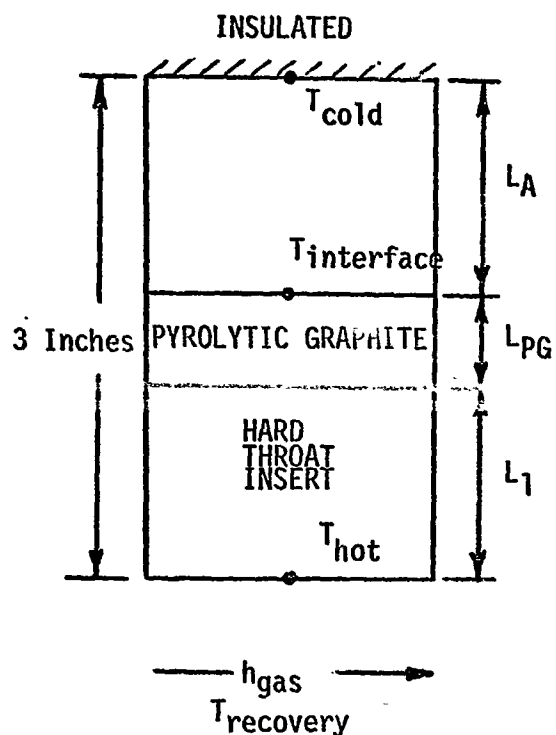
TABLE 4-1 LMDE (QUAL B) DUTY CYCLE

TIME seconds	CHAMBER PRESSURE psi	HEAT TRANSFER COEFFICIENT, $\bar{h}_{\text{gas}}$ Btu/in ² -sec-°F	RECOVERY TEMPERATURE °F
<u>Initial Burn</u>			
0 - 25.999	10	0.17×10^{-3}	3800
26 - 32.999	100	1.05×10^{-3}	3800
<u>Coast</u>			
33 - 3632.999	0	1.0×10^{-12}	0
<u>Final Burn</u>			
3633 - 3658.999	10	0.17×10^{-3}	3800
3659 - 4022.99	100	1.05×10^{-3}	3800
4023 - 4098	60	0.68×10^{-3}	3800
4223	40	0.52×10^{-3}	3800
4233	30	0.41×10^{-3}	3800
4533	20	0.29×10^{-3}	3800
4600	0 (Shutdown)	1.0×10^{-12}	3800

Studies were conducted for the four candidate insert materials: Ta-W, JTA graphite, W-Re/ZrO₂ composite and Pyro-Carb. In addition, studies were made of the effect of inserting a layer of pyrolytic graphite between the insert and the MX2600 ablative as an additional source of high thermal resistance.

Analysis

A sketch of the thermal model used for the analysis is shown below. In the case of the Pyrocarb the material consists of Pyrocarb 400 infiltrated with Si-C. To take this into account in the model, the first 1/10th of an inch of the model is considered to consist of Pyrocarb 751. The thermal operating boundary conditions, gas-side heat transfer coefficients and recovery temperatures are summarized in Table 4-1 for the complete LMDE (Qual B) duty cycle and were obtained from Reference 20 based upon a heat transfer coefficient of 1.5 times the theoretical Bartz coefficient. For the purpose of this analysis, only the first 1600 seconds of the 3600 second coast were investigated since the interface temperature peaks out well within this time period.



For the materials utilized in this analysis a list of the references from which the thermophysical properties were obtained is presented in Table 4-2. The thermal conductivity as a function of temperature for these materials is presented in Figure 4-1 and heating capability as a function of temperature in Figure 4-2.

TABLE 4-2
MATERIAL THERMOPHYSICAL PROPERTIES
REFERENCE TABLE

MATERIAL	PROPERTY REFERENCE
Wire Reinforced Zirconia Composite	TRW Equipment Labs. Cleveland, Ohio
Tantalum/Tungsten	"Thermal Physical Properties of Temperature Solid Materials" Y. S. Touloukian, MacMillan Co., N.Y., 1967
Pyro-Carb 400	Technical Data Bulletin, No. 7.3 Hitco, Gardena, California
Pyro-Carb 751	Private Correspondence from Hitco, Gardena, California
JTA Graphite, Parallel grain Pyrolytic graphite, Perp. grain MX-2600, Parallel Grain	TRW IOC "Thermophysical Property Tables", E. M. Stafford to Distri- bution 4-12-68.

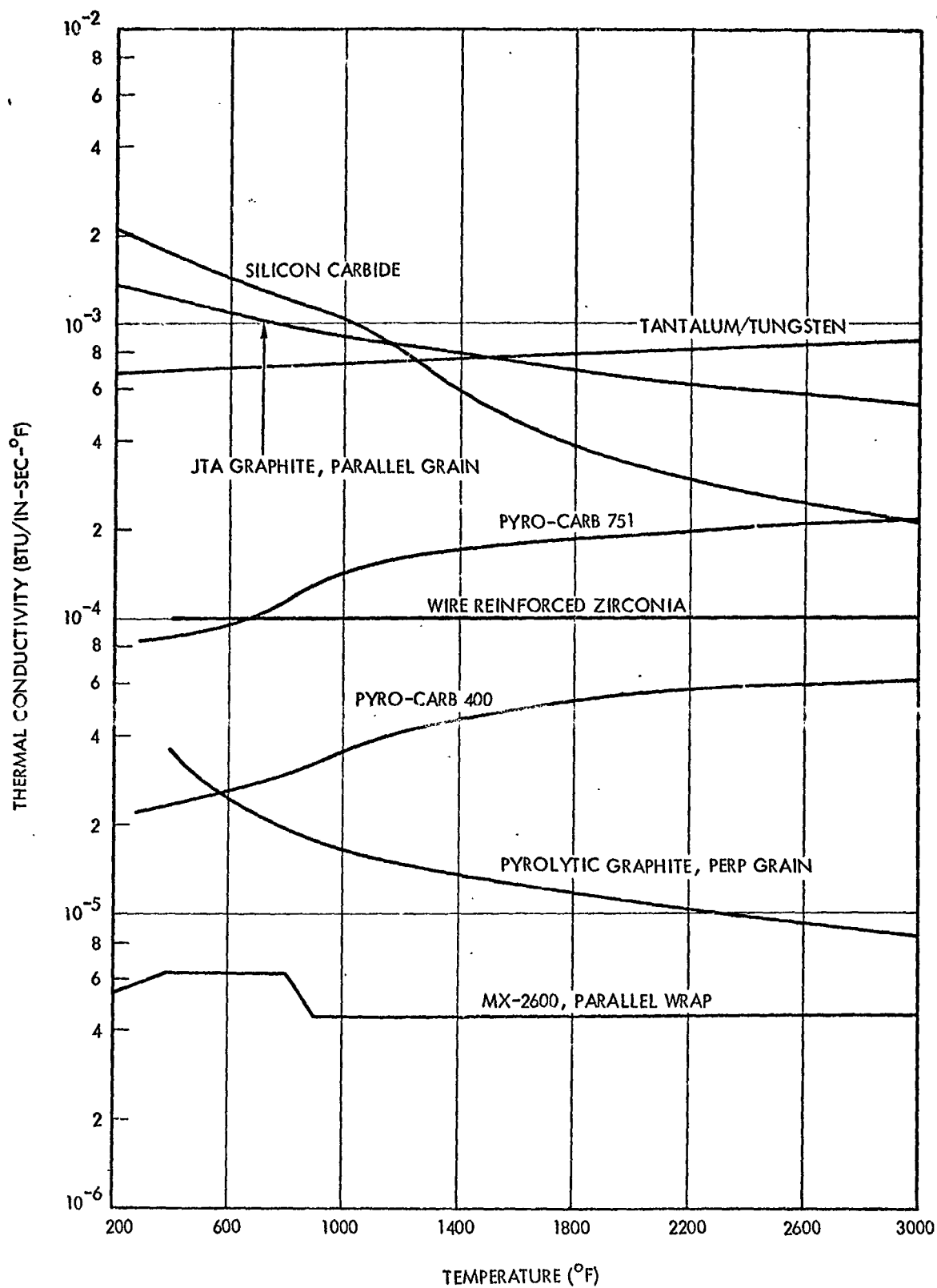


FIGURE 4-1. MATERIAL THERMAL CONDUCTIVITY

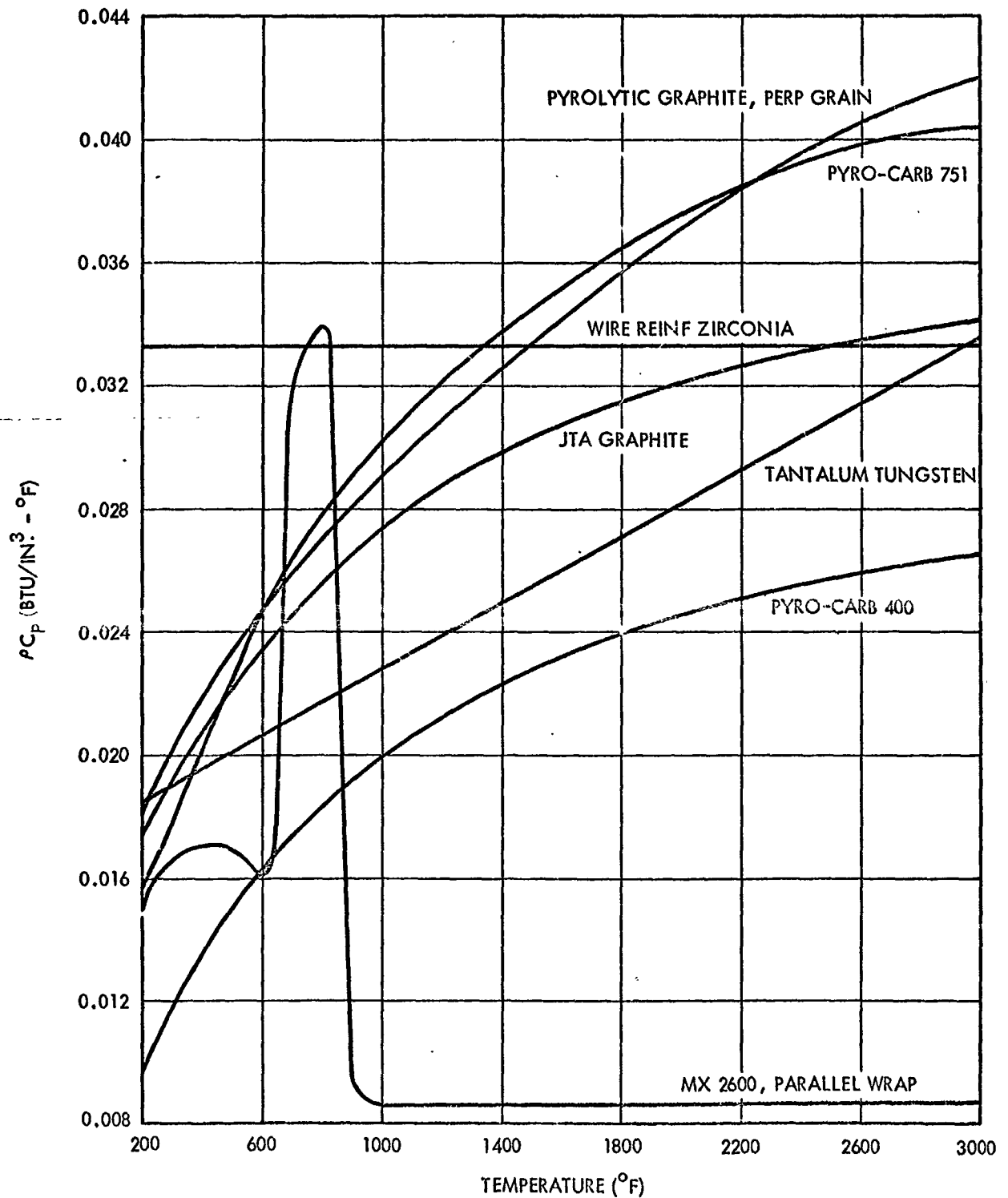


FIGURE 4-2 MATERIAL HEATING CAPABILITY

Results

The results of the parametric analysis of the four selected insert materials are presented in Figures 4-3 through Figure 4-11 and are summarized in Tables 4-3 and 4-4.

- a) Figures 4-3 through 4-6 present the maximum interface temperature during initial firing and soakback for each material as a function of insert thickness and pyrolytic graphite thickness.
- b) Figure 4-7 presents data culled from Figures 4-3 through 4-6 and shows the minimum insert thickness required to maintain the interface temperature below 700°F as a function of insert material and pyrolytic graphite thickness.
- c) Figures 4-8 through 4-11 present temperature-time histories at the interface for each of the four materials. Results are presented for each combination of pyrocarb and pyrolytic graphite thickness.

Discussion

The results of Figure 4-7 are quite important in illustrating two points:

- a) The most favorable insert material from the point of view of maintaining interface temperatures below 700°F is wire reinforced zirconia which showed significantly lower required insert thicknesses than the other materials investigated.
- b) The insertion of a layer of pyrolytic graphite has a pronounced effect upon interface temperature. For three of the materials, 0.3 inches of pyrolytic graphite results in a reduction of 0.5 inches in insert material. Thus there may be a significant trade-off between insert thickness and pyrolytic graphite thickness. In the case of wire reinforced zirconia the trade-off is not quite as favorable since its thermophysical properties more closely approach those of pyrolytic graphite.

TABLE 4-3
MAXIMUM ABLATOR INTERFACE TEMPERATURES DURING COAST
FOLLOWING 33 SECOND FIRING

Material	L_1 (Inches)	L_{PG} (INCHES)					
		L_A	0.0	L_A	0.3	L_A	0.6
JTA	0.5	2.5	1869	2.2	1132	1.9	757
	1.0	2.0	1174	1.7	795	1.4	614
	1.5	1.5	830	1.2	618	0.9	501
W-Re/ZrO ₂	0.5	2.5	1185	2.2	765	1.9	575
	1.0	2.0	567	1.7	461	1.4	387
	1.5	1.5	376	1.2	325	0.9	296
Ta-W	0.5	2.5	2055	2.2	1077	1.9	756
	1.0	2.0	1228	1.7	800	1.4	601
	1.5	1.5	835	1.2	597	0.9	494
Pyro-Carb	0.5	2.5	1627	2.2	960	1.9	674
	1.0	2.0	956	1.7	663	1.4	534
	1.5	1.5	678	1.2	516	0.9	444

TABLE 4-4
MINIMUM INSERT THICKNESS REQUIRED TO MAINTAIN ABLATOR
INTERFACE TEMPERATURE BELOW 700°F
DURING COAST FOLLOWING 33 SECOND FIRING

Material	L_{PG} (Inches)	0.0	0.3	0.6
JTA		~1.8	1.2	0.7
W-Re/ZrO ₂		0.85	0.6	0.2
Ta W		~1.83	1.25	0.67
Pyro-Carb		1.42	0.91	0.43

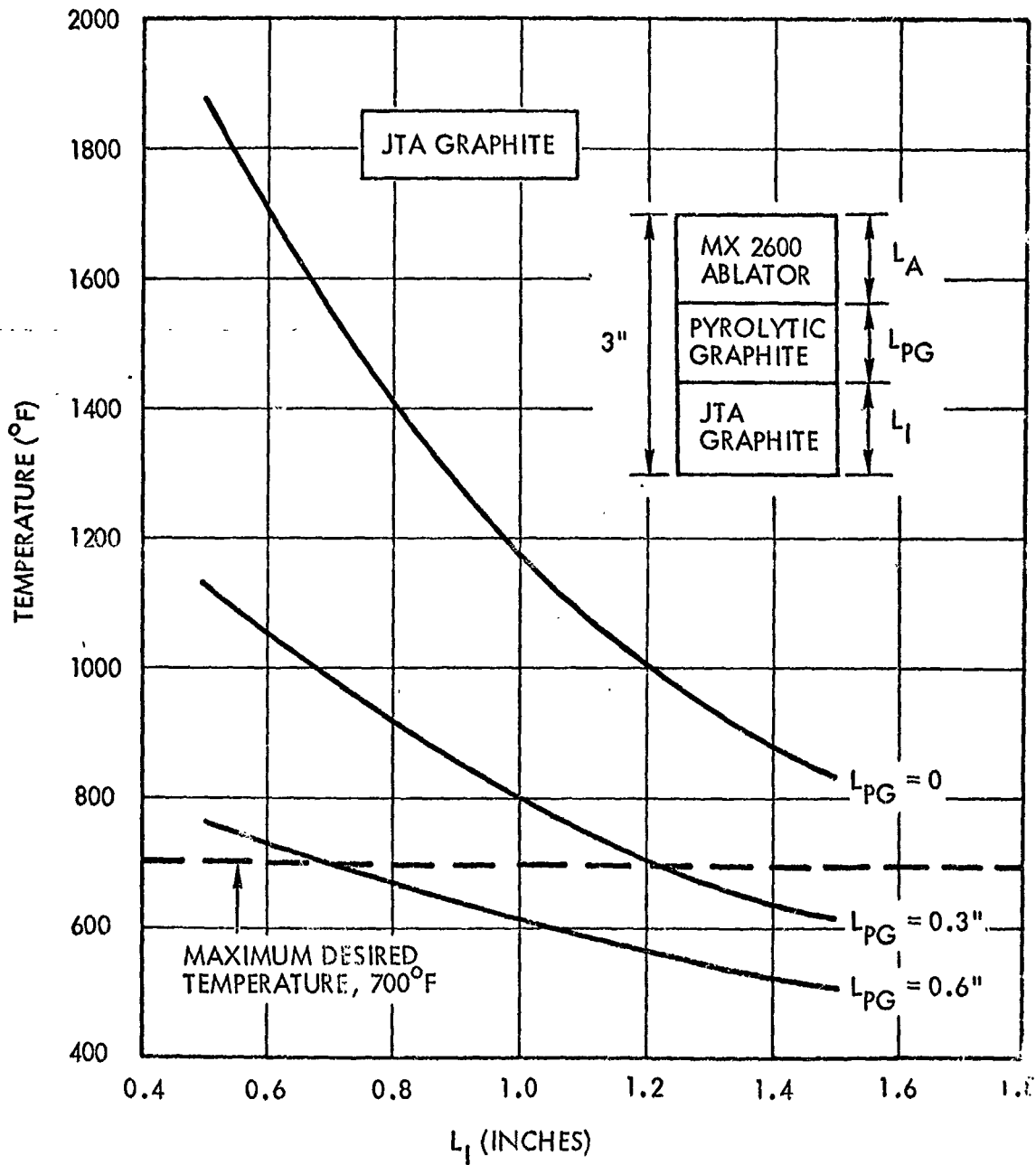
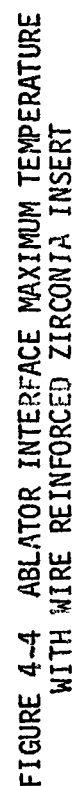


FIGURE 4-3 ABLATOR INTERFACE MAXIMUM TEMPERATURE WITH JTA GRAPHITE INSERT



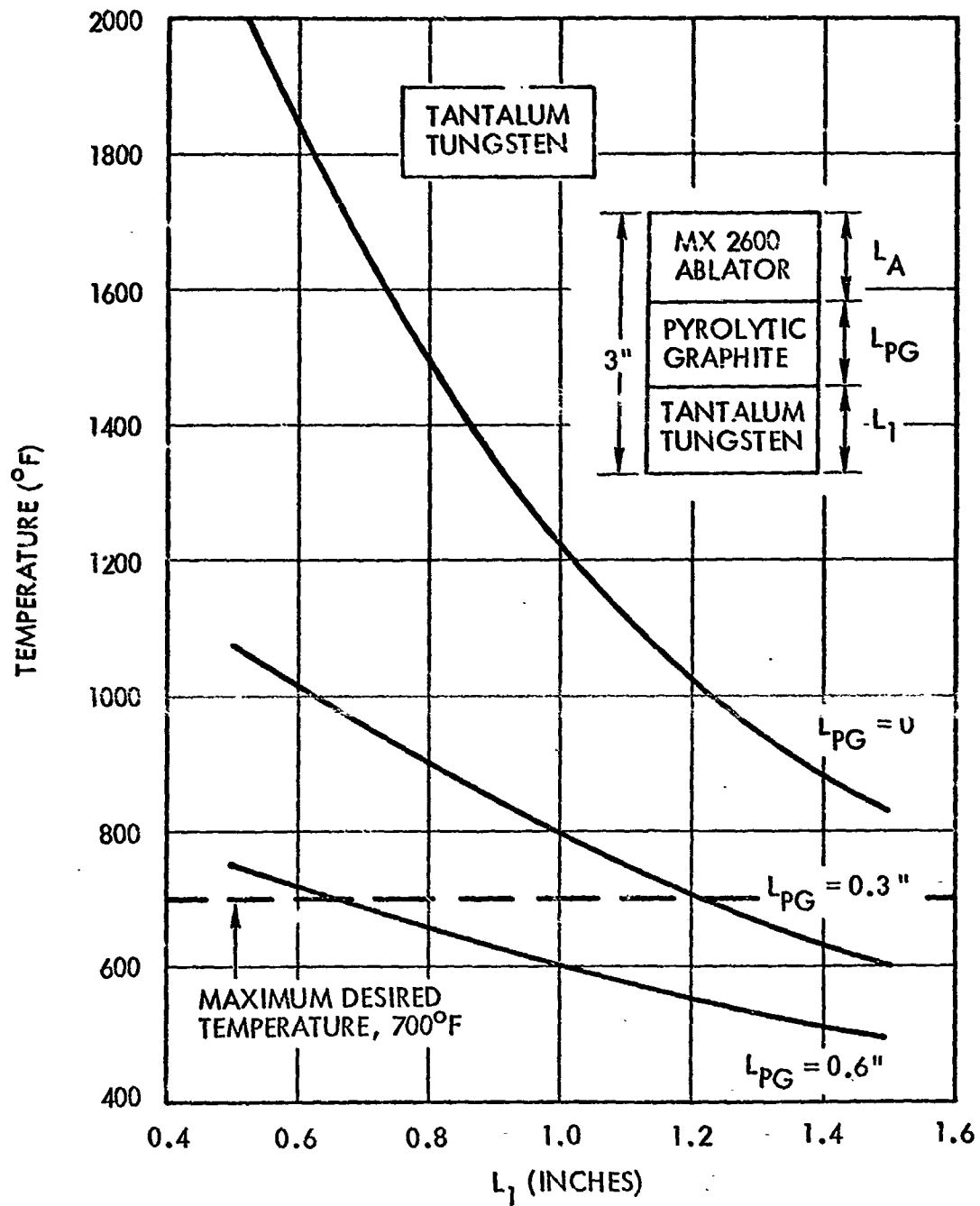


FIGURE 4-5 ABLATOR INTERFACE MAXIMUM TEMPERATURE
WITH TANTALUM-TUNGSTEN INSERT

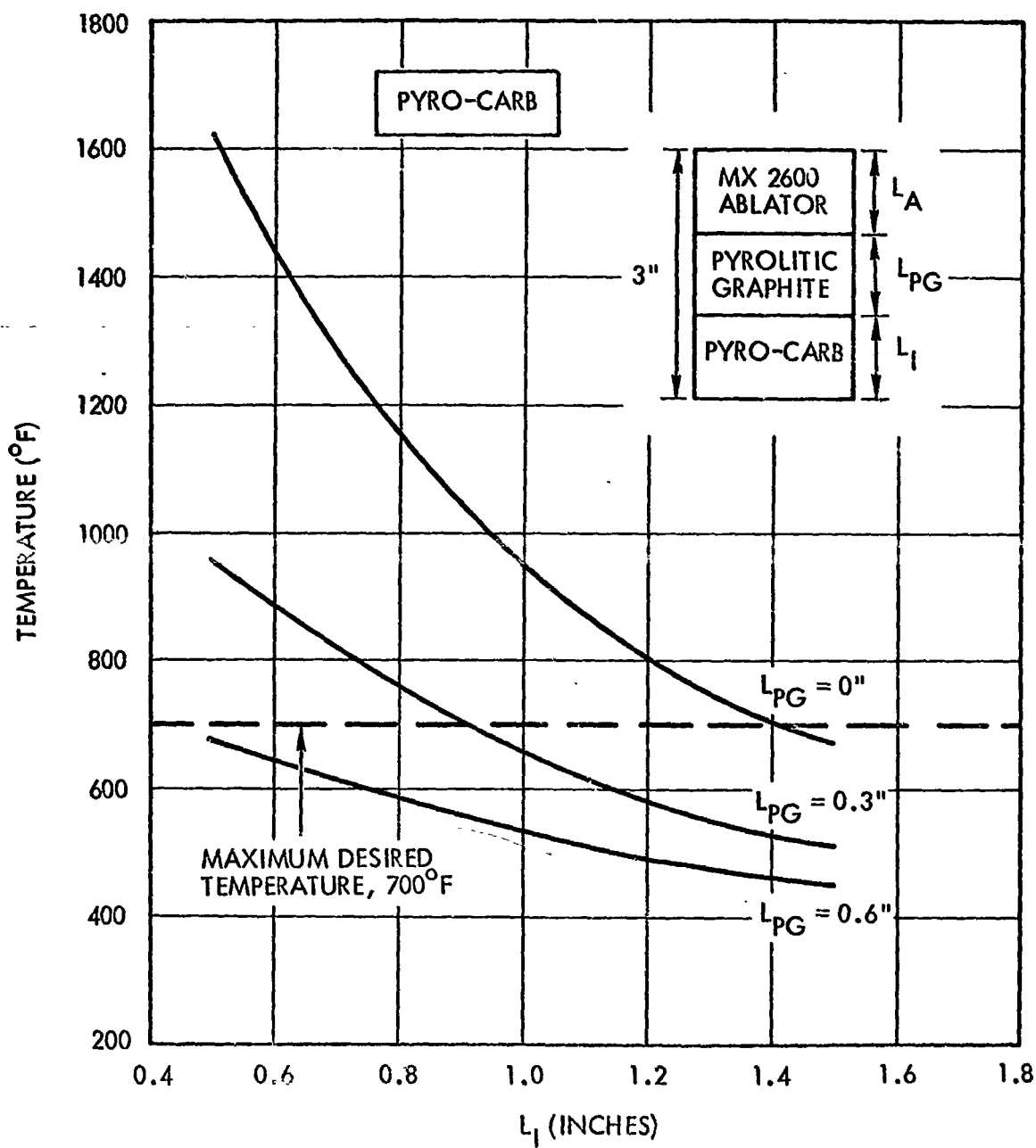


FIGURE 4-6 ABLATOR INTERFACE MAXIMUM TEMPERATURE WITH PYRO-CARB INSERT

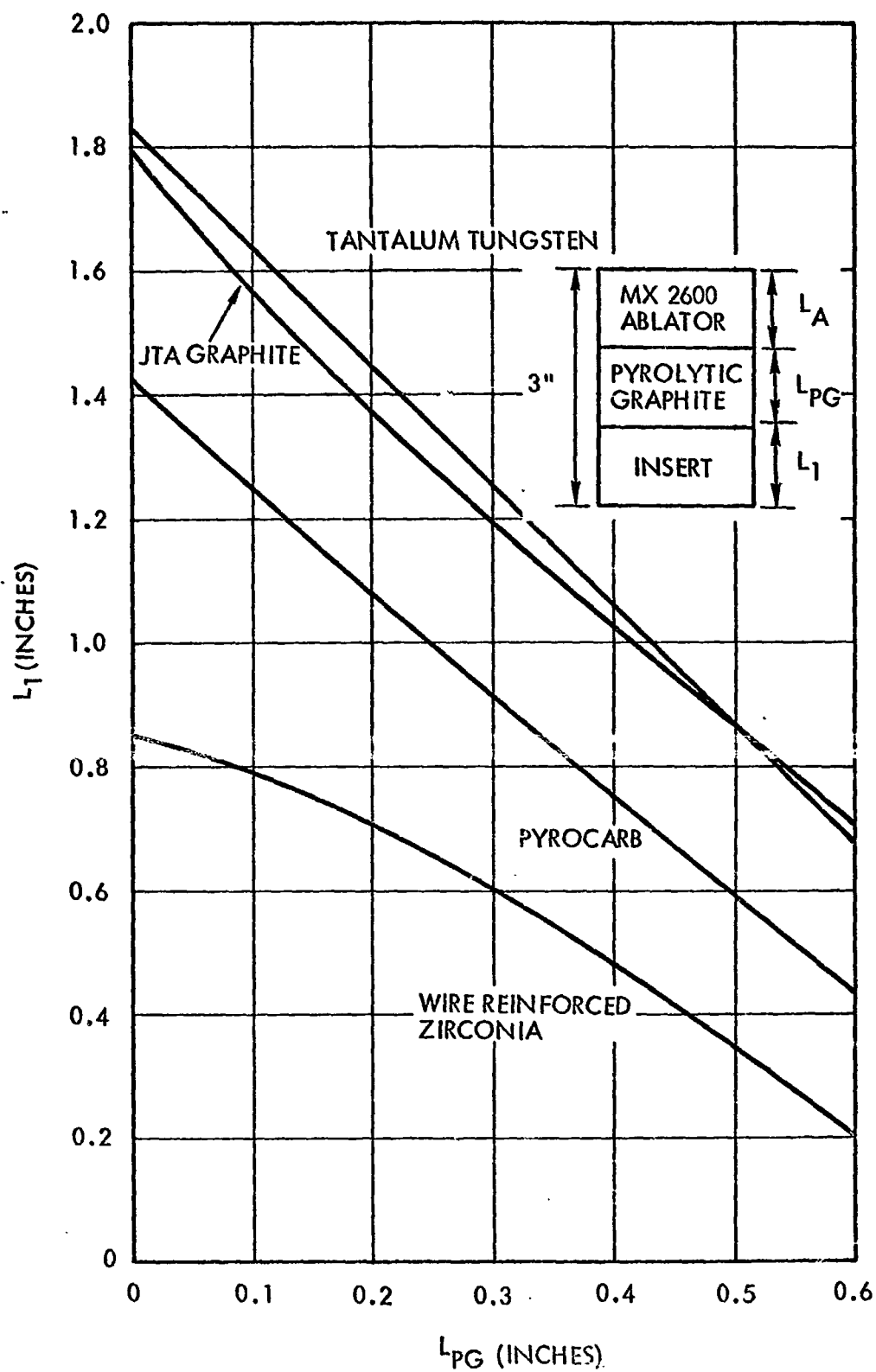


FIGURE 4-7 MINIMUM INSERT THICKNESS REQUIREMENTS
FOR 700°F INTERFACE SOAK TEMPERATURE

The results presented in Figures 4-8 through 4-11 indicate that after the interface has attained its maximum temperature the decay in temperature during the coast is quite slow. This must be borne in mind if insert designs are contemplated where the interface temperature is allowed to go above 700°F. In this case the interface region may remain above this temperature for several hundred seconds.

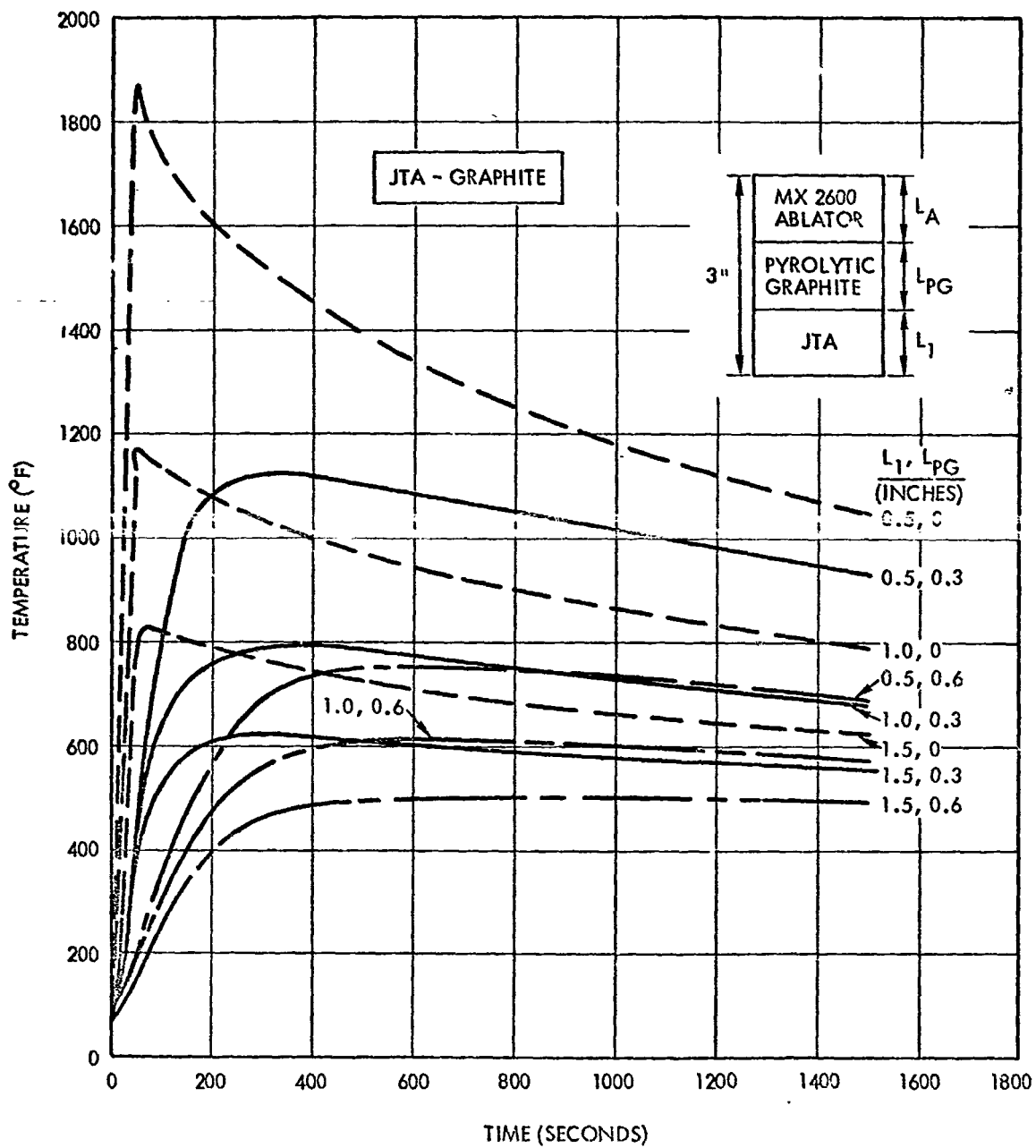


FIGURE 4-8 ABLATOR INTERFACE TEMPERATURE HISTORY DURING INITIAL FIRING AND SOAKBACK WITH JTA GRAPHITE INSERT

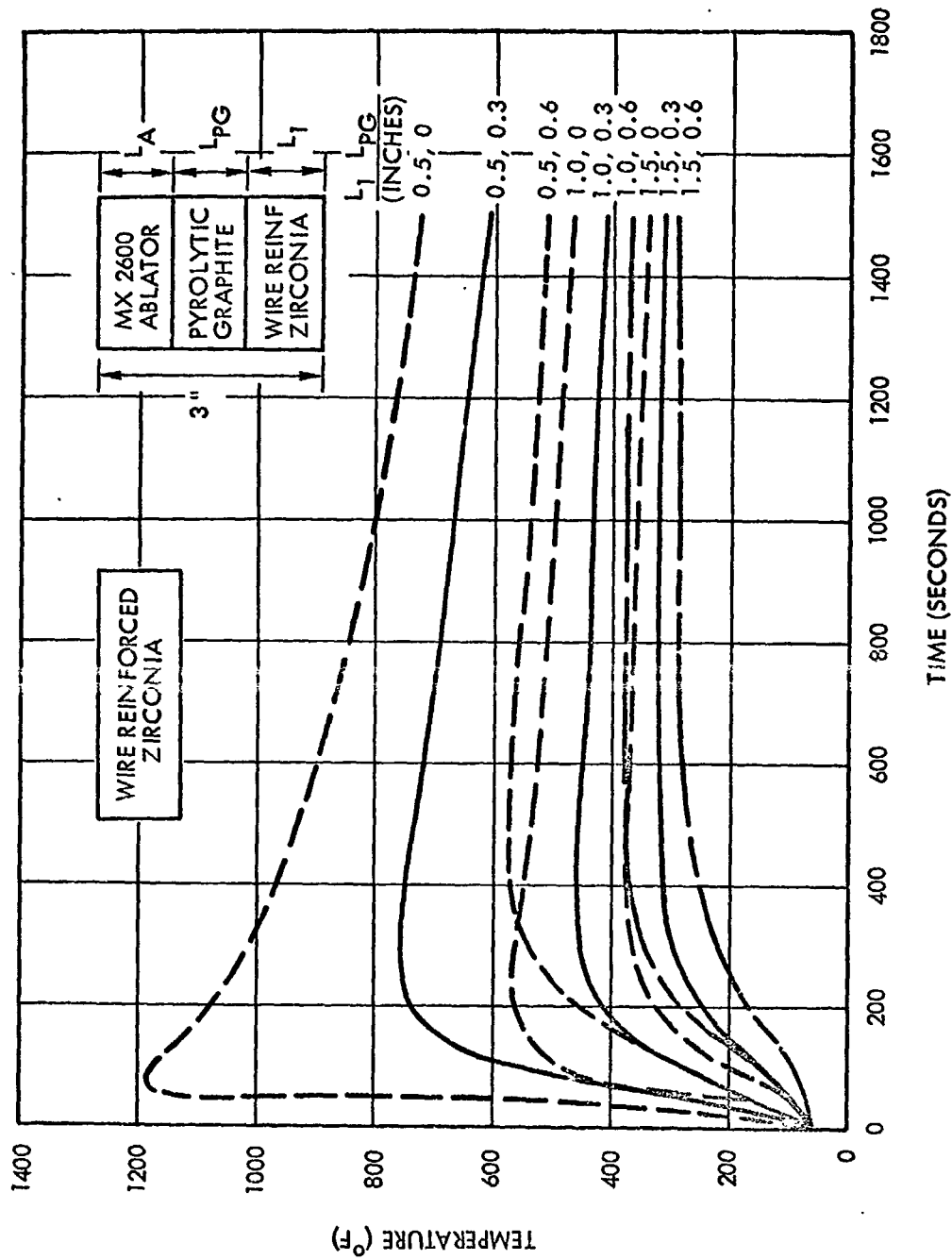


FIGURE 4-9 ABLATOR INTERFACE TEMPERATURE HISTORY DURING INITIAL FIRING AND SOAKBACK WITH WIRE REINFORCED ZIRCONIA INSERT

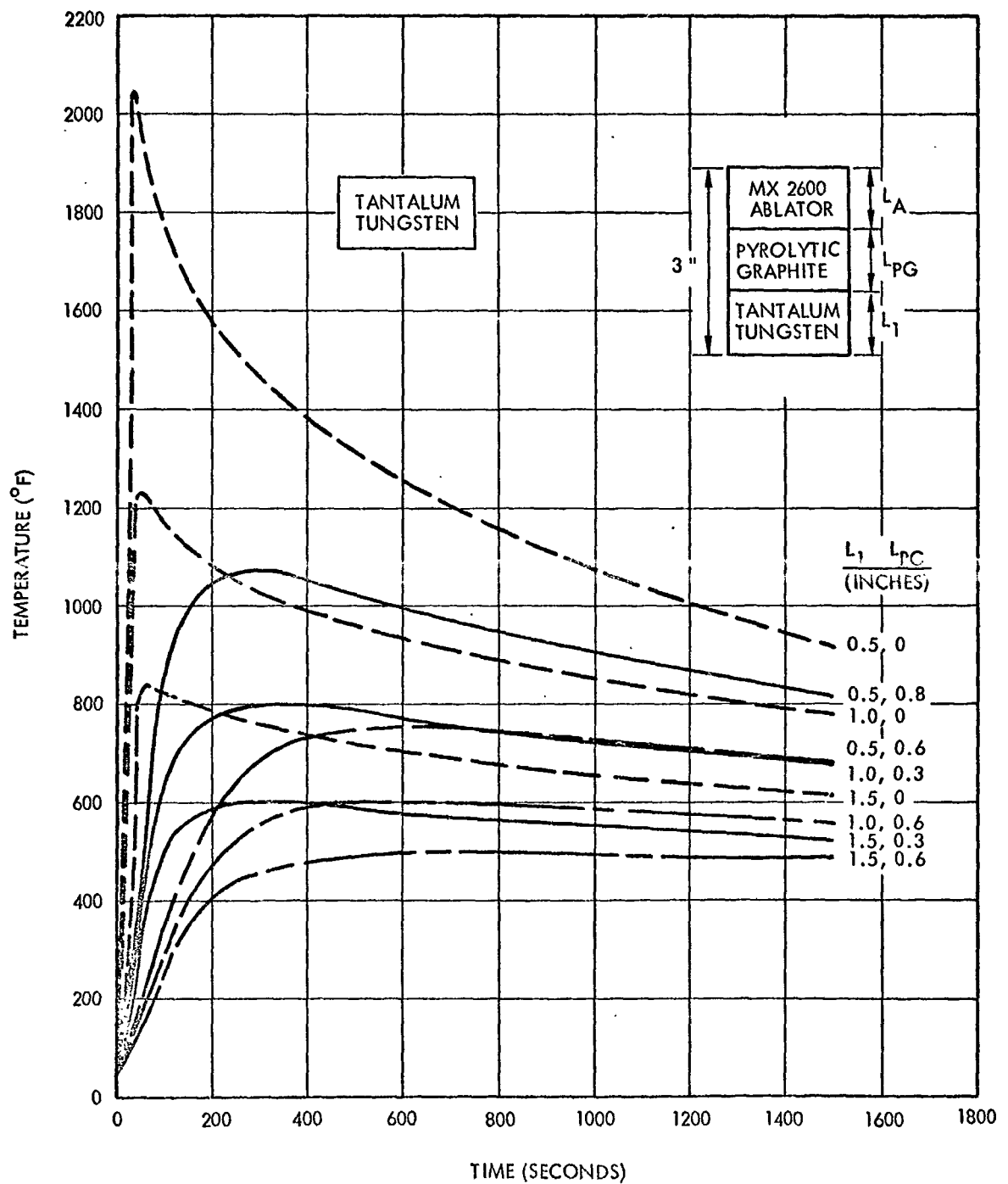


FIGURE 4-10 ABLATOR INTERFACE TEMPERATURE HISTORY DURING INITIAL FIRING AND SOAKBACK WITH TANTALUM-TUNGSTEN INSERT

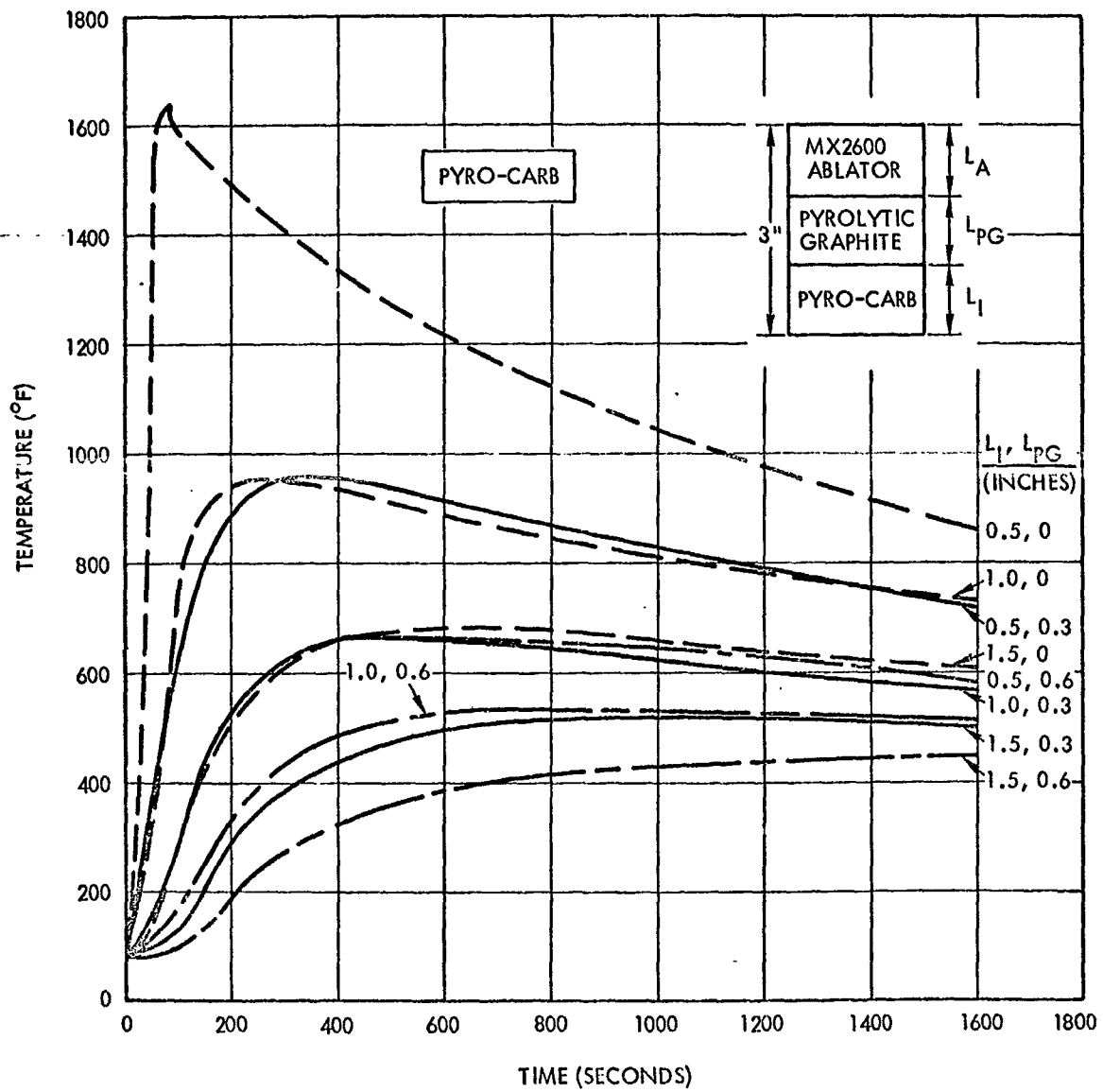


FIGURE 4-11 ABLATOR INTERFACE TEMPERATURE HISTORY DURING INITIAL FIRING AND SOAKBACK WITH PYRO-CARB INSERT

4.1.2 Insert Thermal Analysis

Thermal Models

The thermal models shown in Figure 4-12 were analyzed on a one dimensional basis with radial conduction considered. The thermal operating boundary conditions, gas side heat transfer coefficients, and recovery temperatures were summarized in Table 4-1 for the LMDE (Qual B) duty cycle. In order to minimize computation time, the one hour coast soakback was not simulated. Instead, the initial temperatures for the final burn were obtained from Section 4.1.1 interface studies.

Results

The results of the parametric analyses of the four selected insert materials are presented individually in Figures 4-13 through 4-24 and are summarized in Tables 4-5 and 4-6. Basically, three sets of results were obtained for each insert material and are as follows: (1) backwall and interface temperatures at the end of the mission duty cycle (TIME = 4600 Sec.) as a function of insert thickness and total thickness, (2) maximum insert temperature gradients at the end of the initial burn (TIME = 33 Sec.) and (3) typical time - temperature histories during the initial and final burns.

For each of the four insert materials a brief description of the physical configuration and results that were obtained from the analysis are as follows:

Pyro-Carb - Figure 4-12 shows the model used in the thermal analysis for the Pyro-Carb insert. MX-2600 parallel wrap was used for the insulating material; however pyrolytic graphite was not included in the analysis. The Pyro-Carb insert will actually consist of Pyro-Carb infiltrated with Si-C. Due to lack of thermal data on the impregnated material, the analysis assumed a composite broken down into a 0.25 inch layer of Si-C backed by non-infiltrated Pyro-Carb. The results for the Pyro-Carb are presented in Figures 4-13 through 4-15.

JTA - The thermal model for the JTA insert is shown in Figure 4-12. The insulating material was MX2600, 60° centerline oriented and pyrolytic graphite thicknesses up to 0.3 inches were investigated.

ONE DIMENSIONAL
RADIAL CONDUCTION CONSIDERED

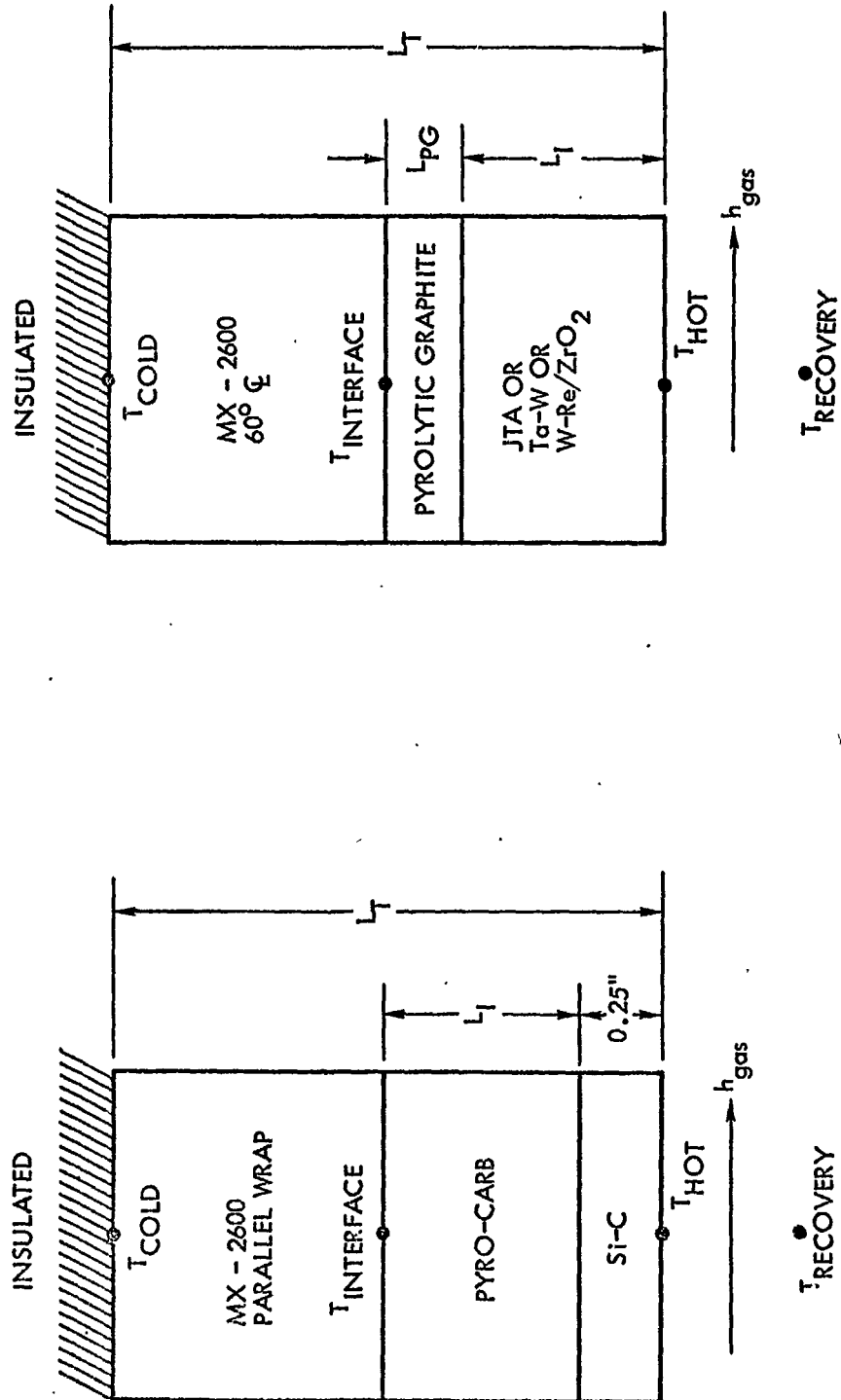


FIGURE 4-12 THERMAL ANALYSIS MODELS

TABLE 4-5 SUMMARY, LMDE HARD THROAT INSERT THERMAL ANALYSIS

MATERIAL	L_J INCHES	BACKWALL TEMP. T_{Cold} , °F				INTERFACE TEMP. °F	INSERT THERMAL GRADIENTS °F	
		L_T , Inches $L_{PG} = 0.0"$						
		2	3	4		$L_T = 2 \text{ to } 4"$ $L_{PG} = 0.0"$	$L_T = 2 \text{ to } 4"$ $L_{PG} = 0.0"$	
JTA	0.5	1300	200	100		3575	2840	400
	1.0	2500	575	300		3625	2950	1120
	1.5	3400	1000	550		3675	3050	1440
W-Re/ZrO ₂	0.5	750	100	100		3600	2650	2300
	1.0	1625	225	100		3425	2575	2850
	1.5	2100	350	120		2870	2100	3000
Ta-W	0.2	750	475	250		3550	2800	100
	0.6	1450	530	250		3650	2825	550
	1.0	2550	600	275		3680	2960	1000
Pyro-Carb	0.25	750	390	235		3480	Not Analyzed	1775
	0.50	880	430	235		3430		2250
	0.75	1030	450	235		3350		2375

TABLE 4-6 THERMAL DESIGN CRITERION

MATERIAL	L _T INCHES	BACKWALL TEMP. $\leq 800^{\circ}\text{F}$ L _T , Inches	INTERFACE TEMP. $\leq 3000^{\circ}\text{F}$ L _T , Inches	
		With L _P G = 0.0"	L _P G = 0.0"	L _P G = 0.3"
JTA	0.5	>2.3	* Temp. >3000	>1.6
	1.0	>2.8	Temp. >3000	>2.6
	1.5	>3.3	Temp. >3000	>3.4
W-Re/ZrO ₂	0.5	>2.0	Temp. >3000	Temp. <3000
	1.0	>2.4	Temp. >3000	Temp. <3000
	1.5	>2.5	L _T > 2.2	Temp. <3000
Ta-W	0.2	>1.9	Temp. >3000	>1.6
	0.4	>2.5	Temp. >3000	>1.8
	0.6	>2.7	Temp. >3000	>2.8
Pyro-Carb	0.25	>1.9	Temp. >3000	Not Analyzed
	0.50	>2.1	Temp. >3000	
	0.75	>2.2	Temp. >3000	

* Interface Temperature was always greater than 3000°F for the wall thicknesses (L_T) that were studied. The exception is shown for W-Re/ZrO₂.

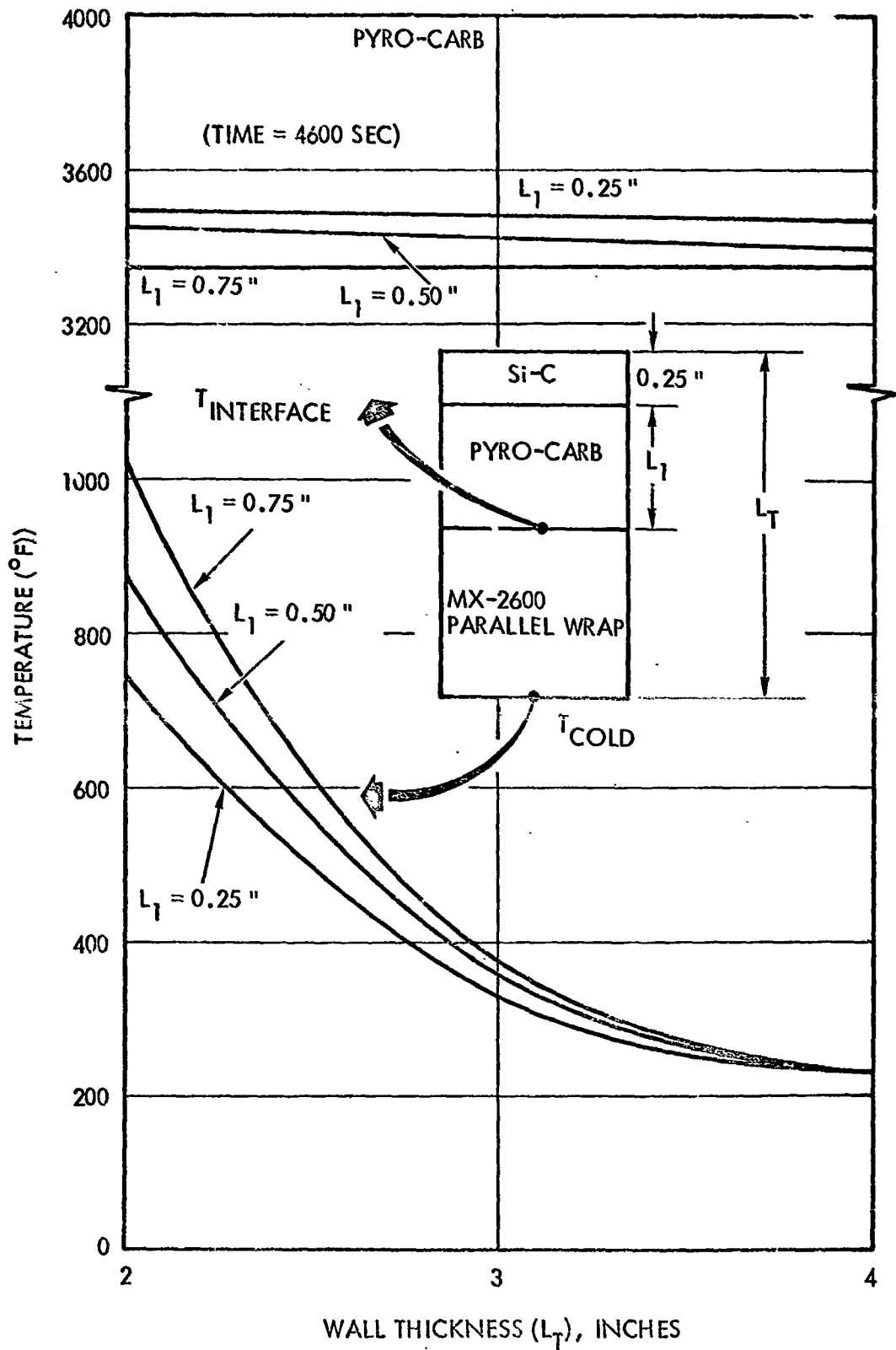


FIGURE 4-13 BACKWALL MATERIAL TEMPERATURES AT THE END OF LMDE DUTY CYCLE WITH PYRO-CARB INSERT

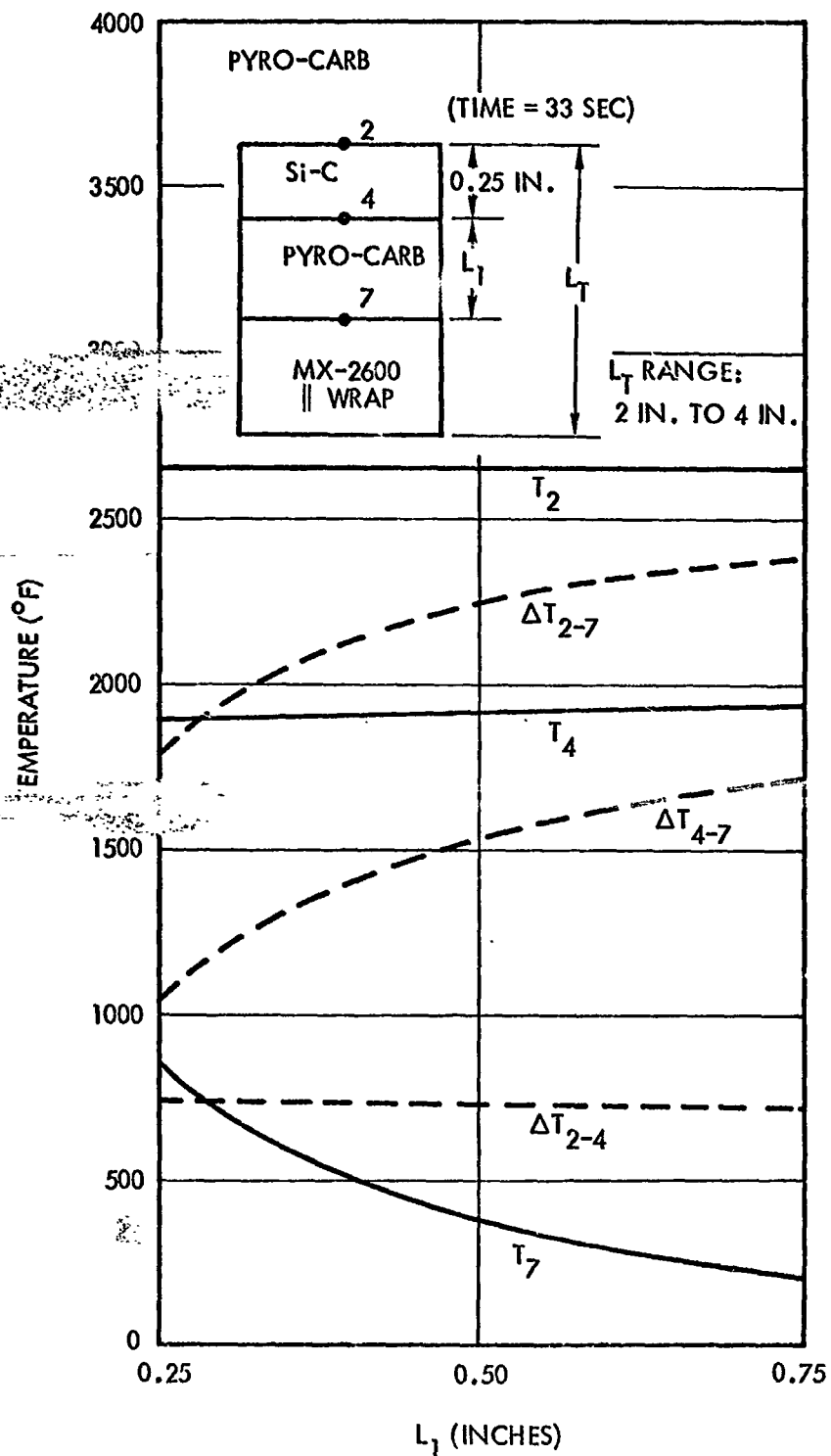


FIGURE 4-14 INITIAL BURN MAXIMUM TEMPERATURE GRADIENTS WITH PYRO-CARB INSERT

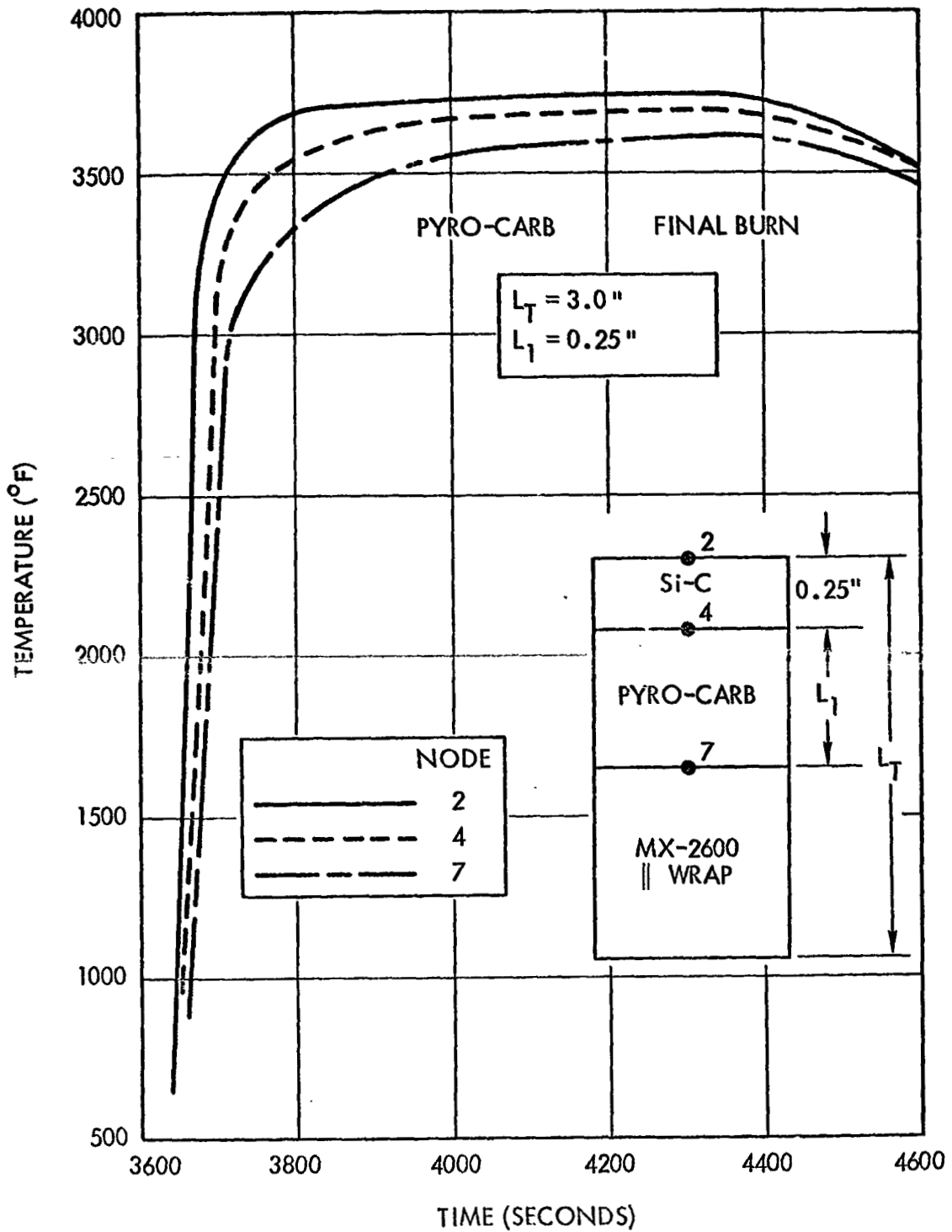


FIGURE 4-15 TYPICAL TIME-TEMPERATURE HISTORY
CURVE WITH PYRO-CARB, FINAL BURN

Figures 4-16 through 4-20 show the results for the JTA. Maximum insert temperature gradients for an initial full thrust condition are shown in Figure 4-18.

W-Re/ZrO₂ - The thermal model for the W-Re/ZrO₂ was the same as for the JTA. The results for the W-Re/ZrO₂ are shown in Figures 4-21 and 4-22.

Ta-W - The thermal model for the Ta-W was the same as for the JTA. The results for the Ta-W are shown in Figures 4-23 and 4-24.

Discussion

In general, the insert thermal gradients, interface temperatures and surface operating temperatures were found to be independent of overall wall thickness and highly dependent upon insert material thickness. The effect of insert material thermal conductivity on interface and back-wall temperatures was found to be relatively proportional to the respective thermal conductivities. The lower conductivity of the W-Re/ZrO₂ composite and the Pyro-Carb insert resulted in lower interface and backwall temperatures; however, the lower conductivities caused more severe insert thermal gradients.

For the material thicknesses studied, the interface temperature could only be maintained less than 3000°F with the inclusion of 0.3 inches of Pyrolytic Graphite. Although the analysis conducted for the Pyro-Carb did not include the Pyrolytic Graphite, the analyses of the other three insert materials indicate that the inclusion of 0.3 inches of Pyrolytic Graphite with the Pyro-Carb would probably have resulted in a 800°F lower interface temperature and had little or no effect upon backwall temperatures and insert thermal gradients.

The significant conclusions derived from the analysis are as follows:

- Backwall temperatures at the end of the mission duty cycle can be maintained below 800°F (with any of the inserts) if total thickness exceeds 2 inches.
- In order to maintain interface temperatures below 3000°F, at least 0.3 inches of pyrolytic graphite is required.

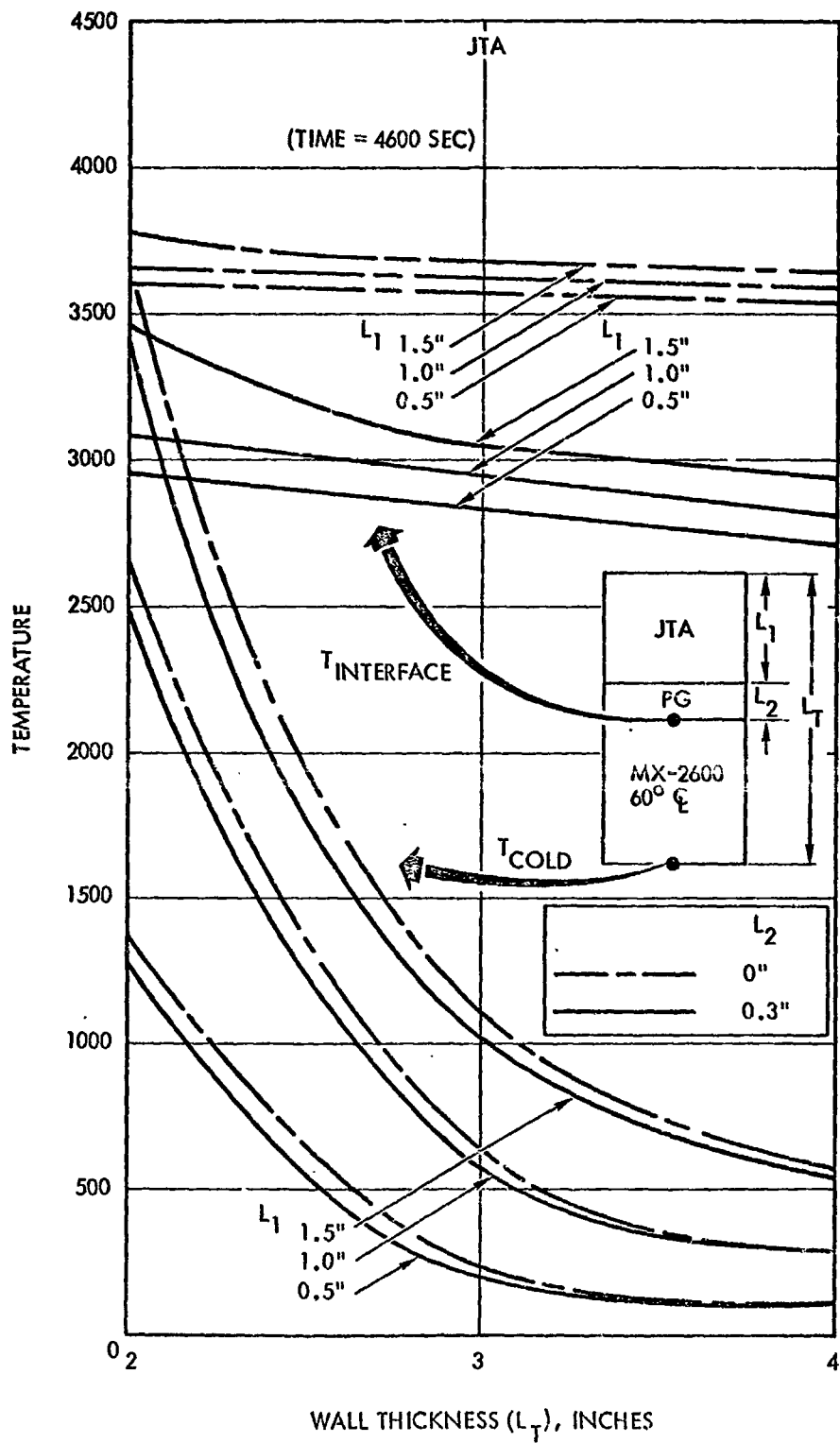


FIGURE 4-16 BACKWALL MATERIAL TEMPERATURES AT THE END OF LMDE DUTY CYCLE WITH JTA INSERT

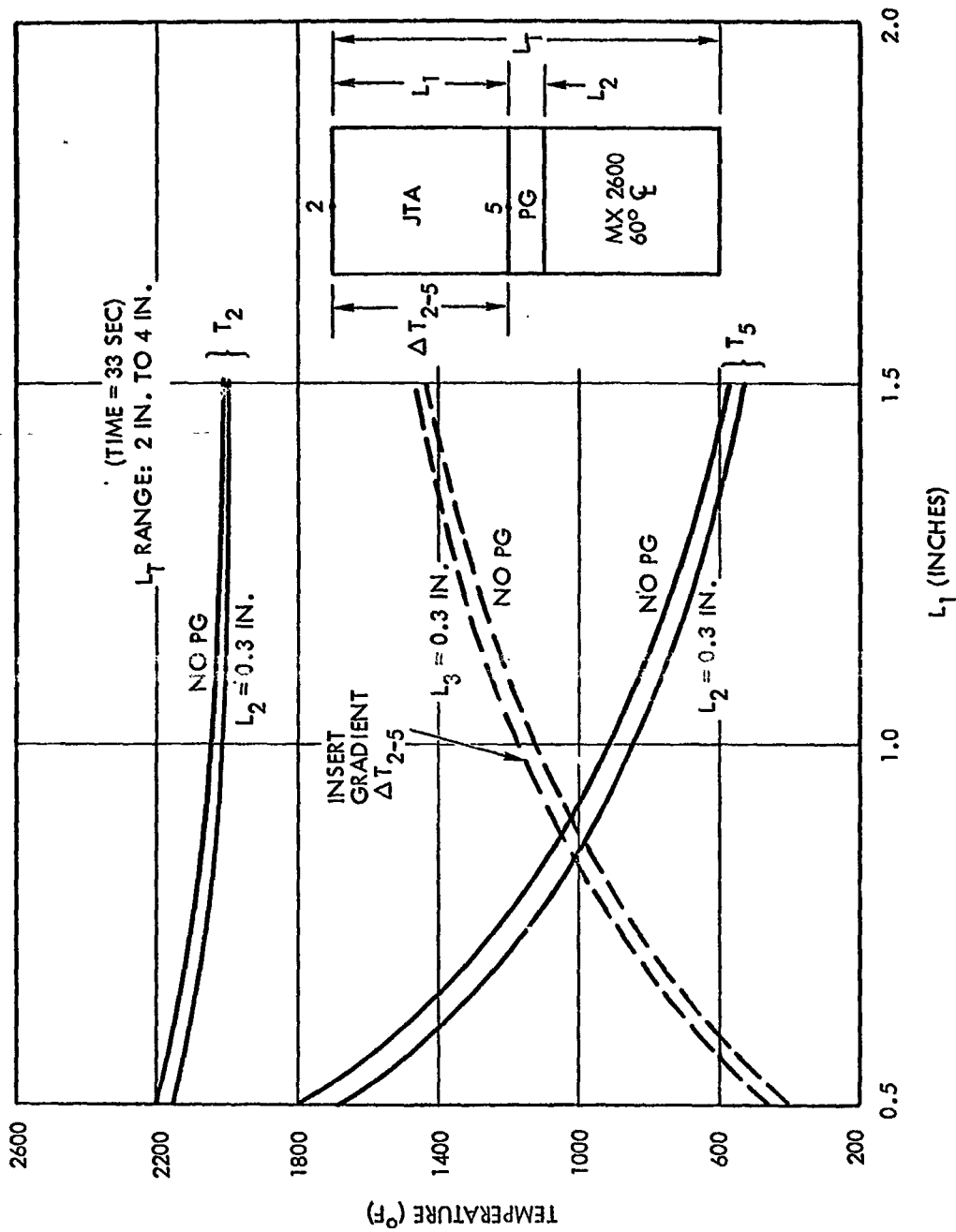


FIGURE 4-17 INITIAL BURN MAXIMUM TEMPERATURE GRADIENTS WITH JTA INSERT

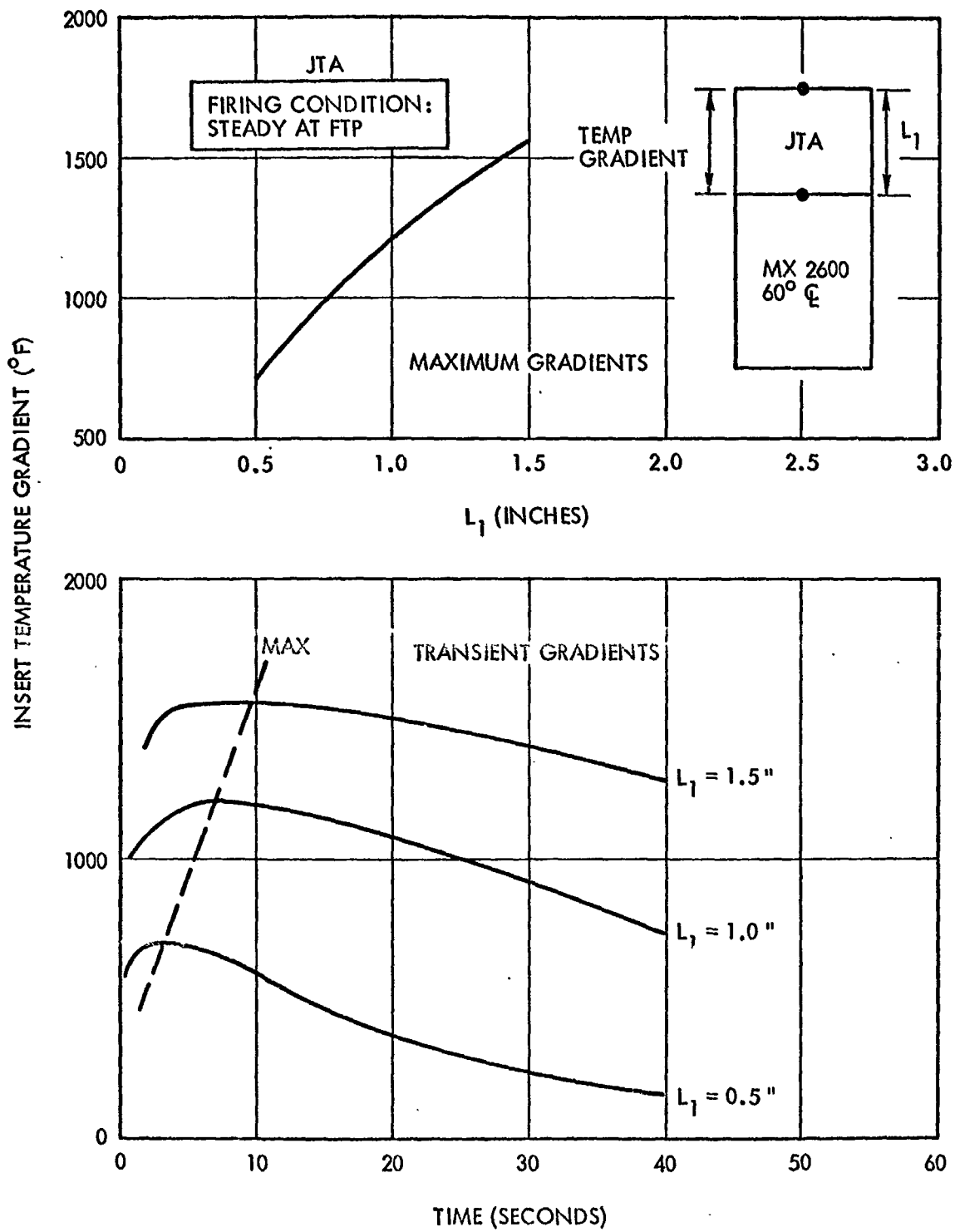


FIGURE 4-18 MAXIMUM GRADIENTS AT FTP, JTA INSERT

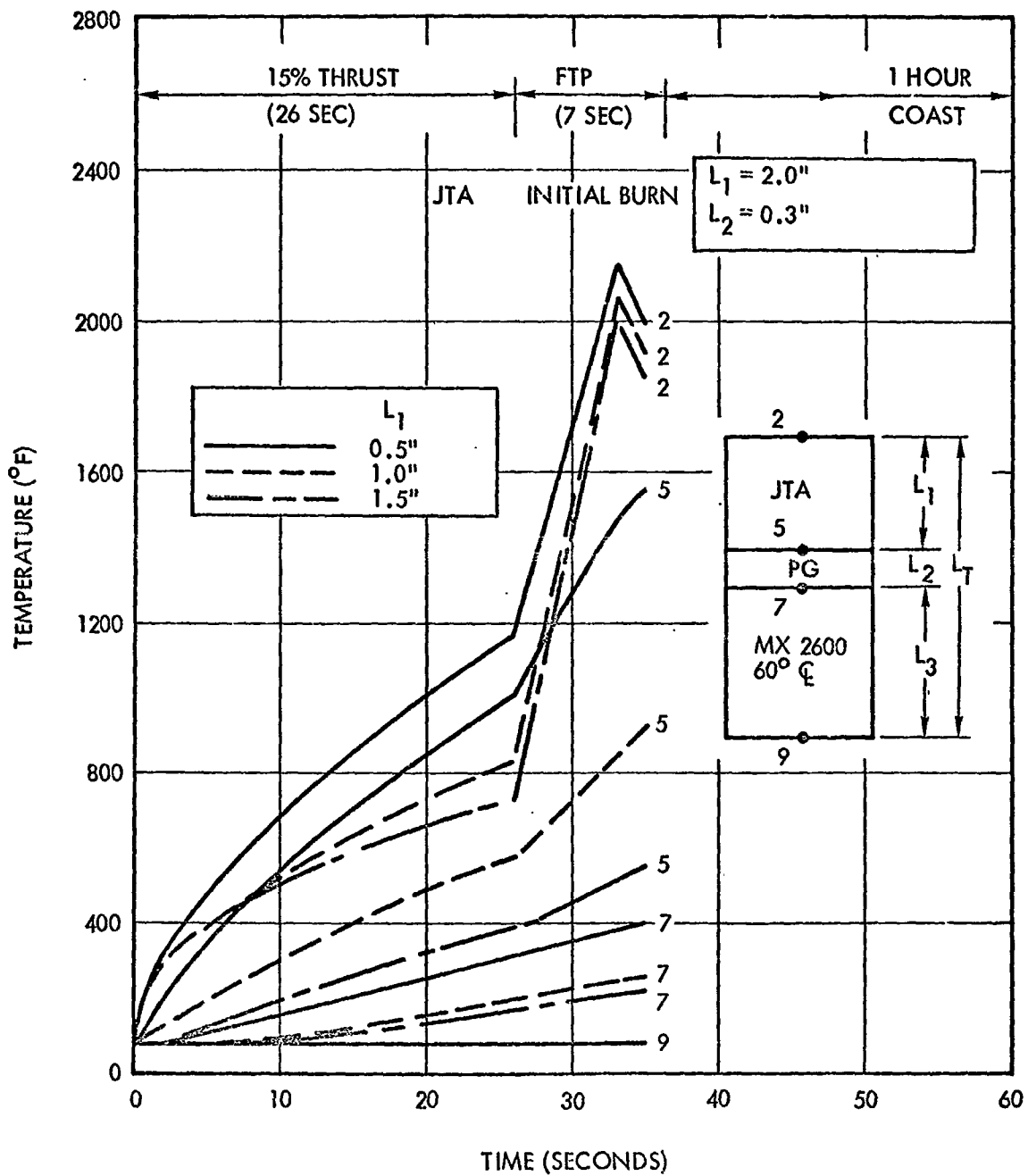


FIGURE 4-19 TYPICAL TIME-TEMPERATURE HISTORY, INITIAL BURN, JTA INSERT

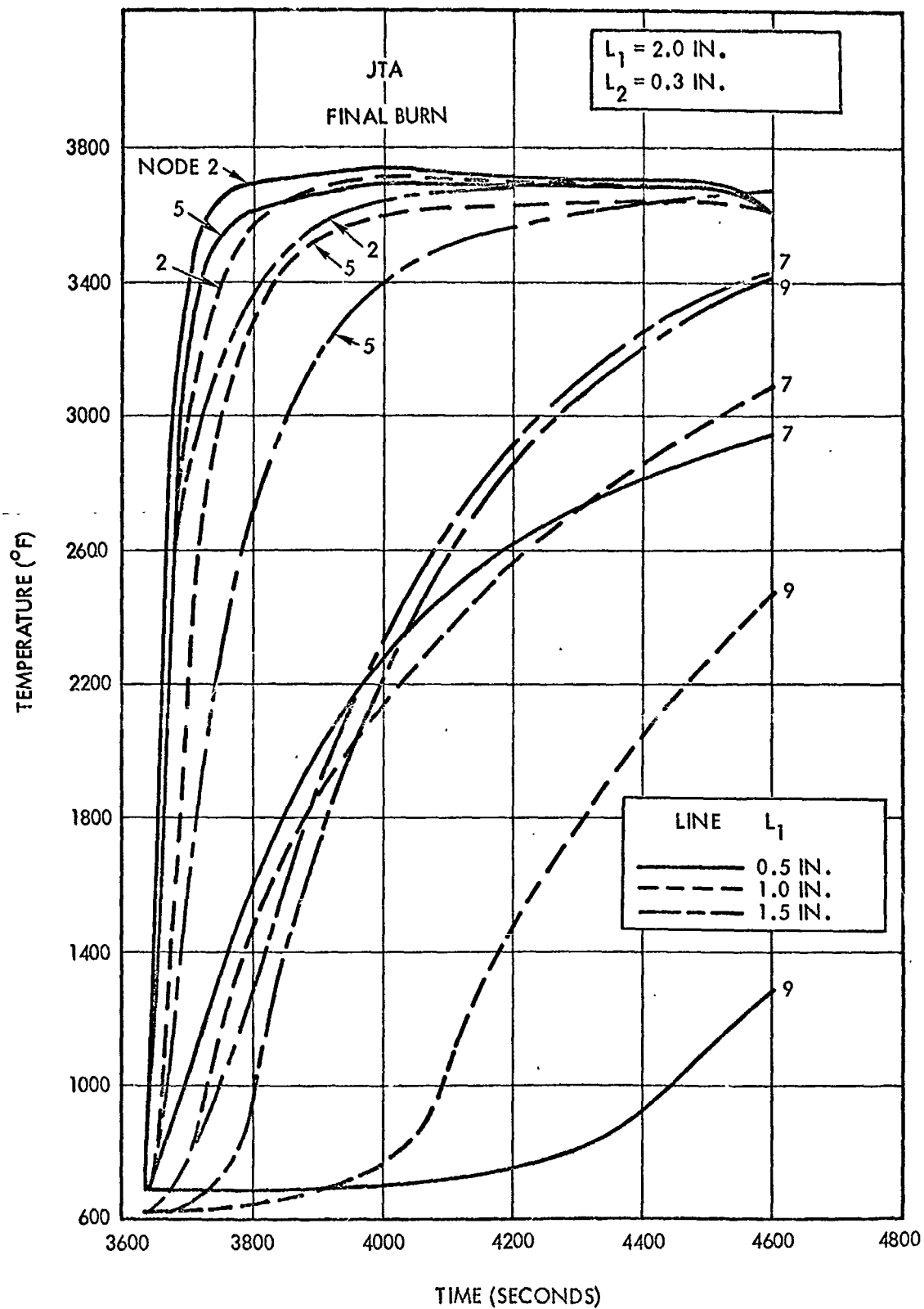


FIGURE 4-20 TYPICAL TIME-TEMPERATURE HISTORY, FINAL BURN, JTA INSERT

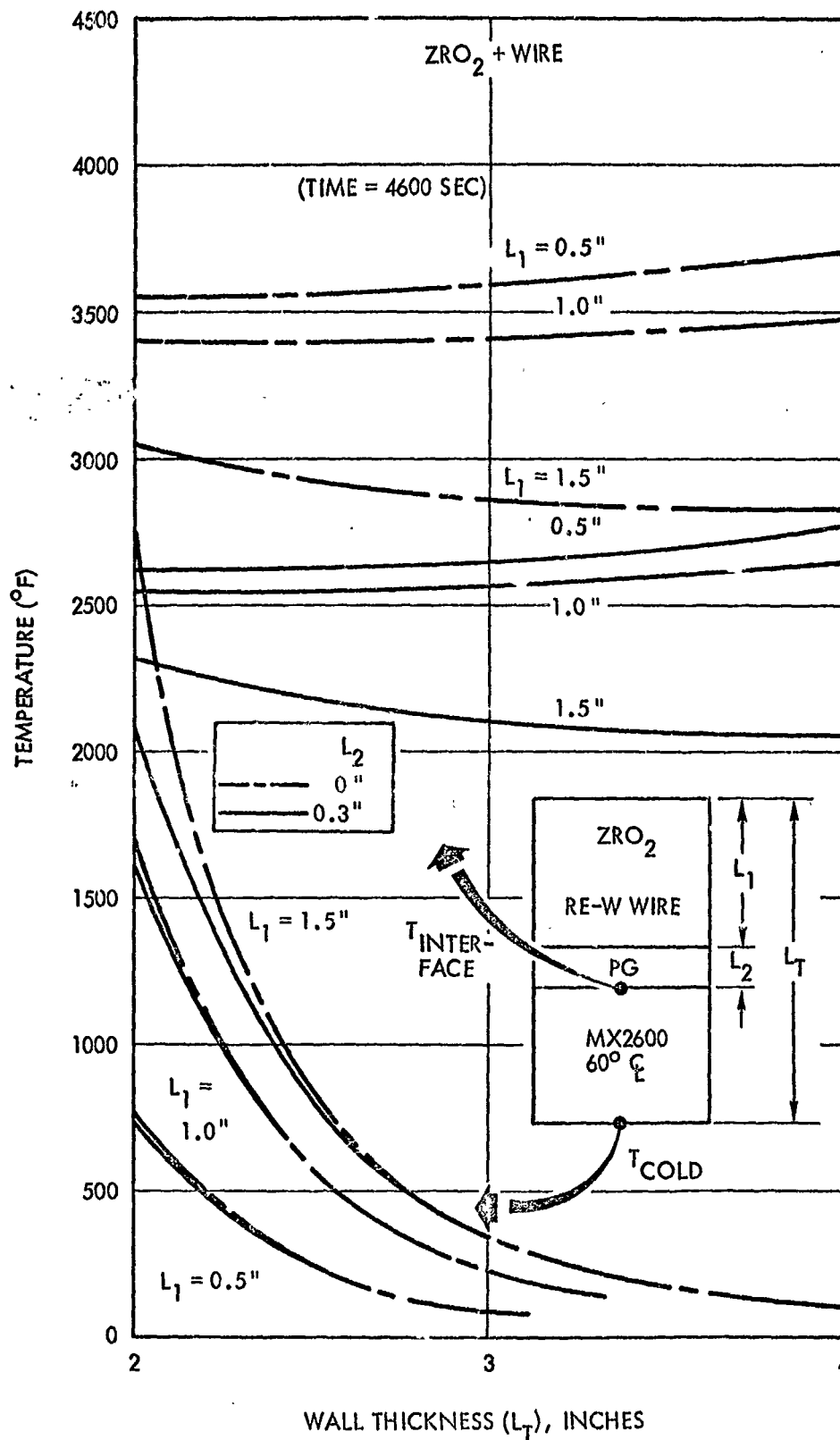


FIGURE 4-21 BACKWALL MATERIAL TEMPERATURES AT THE END OF LMDE DUTY CYCLE WITH WIRE-REINFORCED ZIRCONIA INSERT

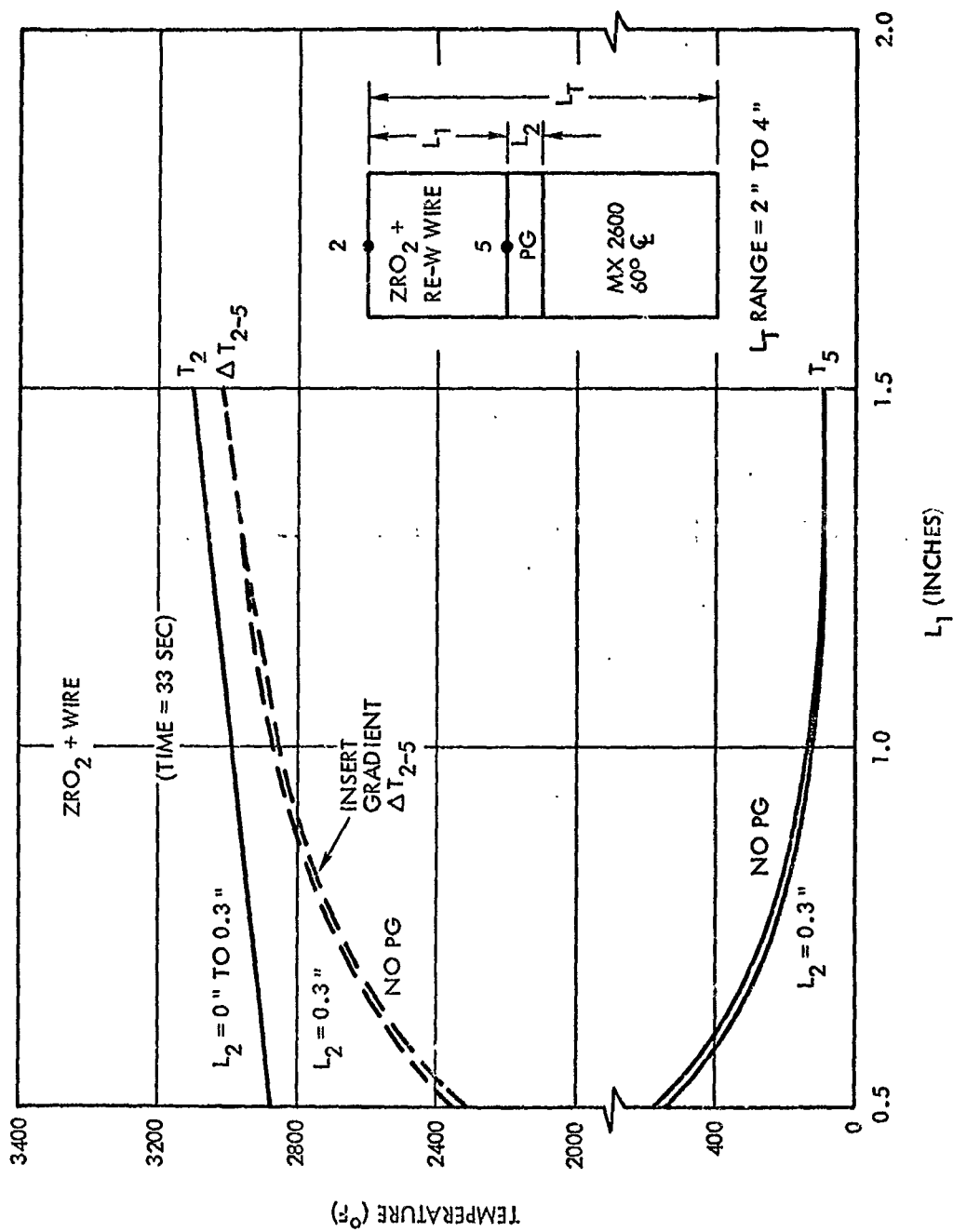


FIGURE 4-22 INITIAL BURN MAXIMUM TEMPERATURE GRADIENT WITH WIRE-REINFORCED ZIRCONIA INSERT

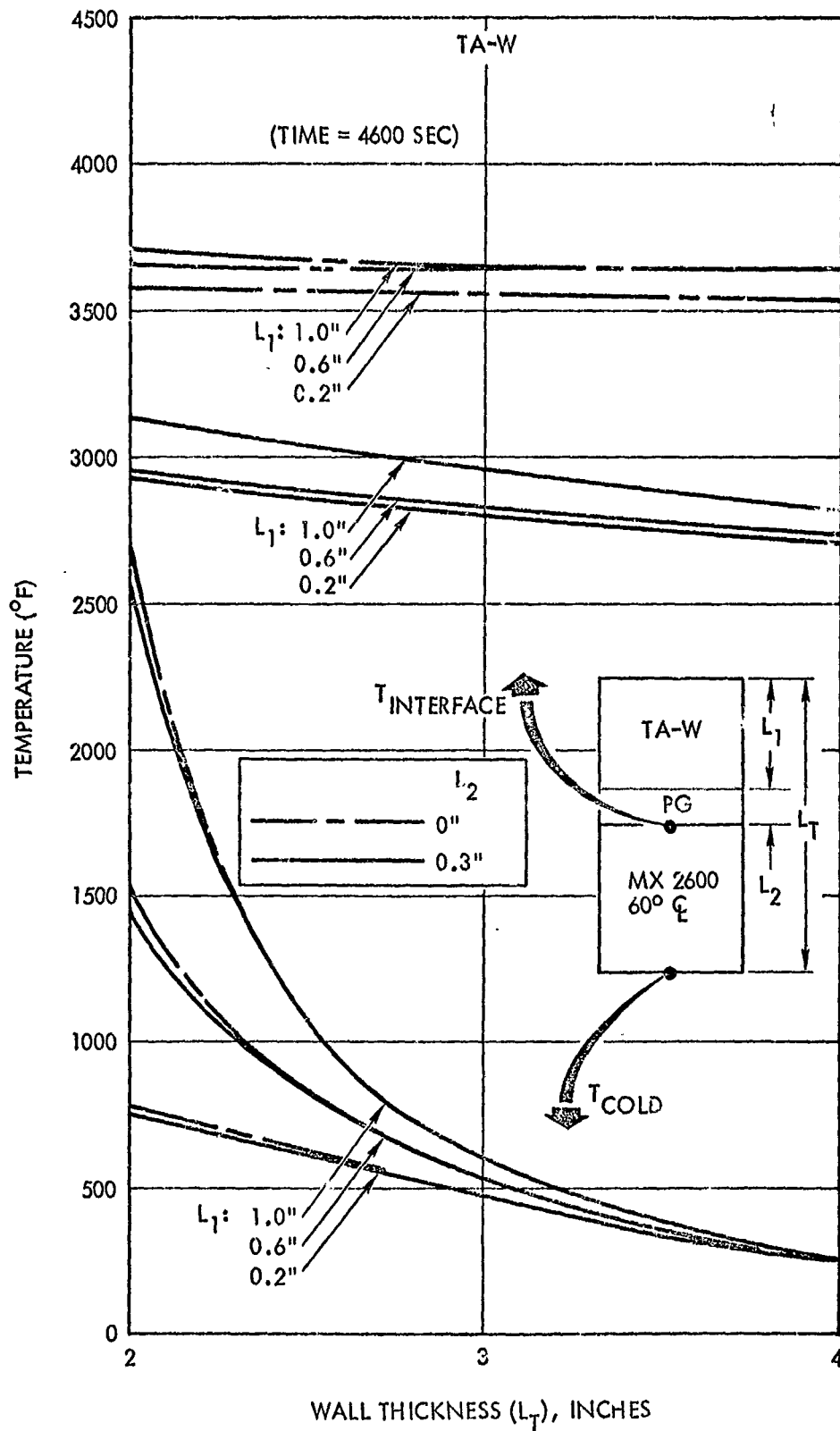


FIGURE 4-23 BACKWALL MATERIAL TEMPERATURES AT THE END OF LMDE DUTY CYCLE WITH TANTALUM TUNGSTEN INSERT

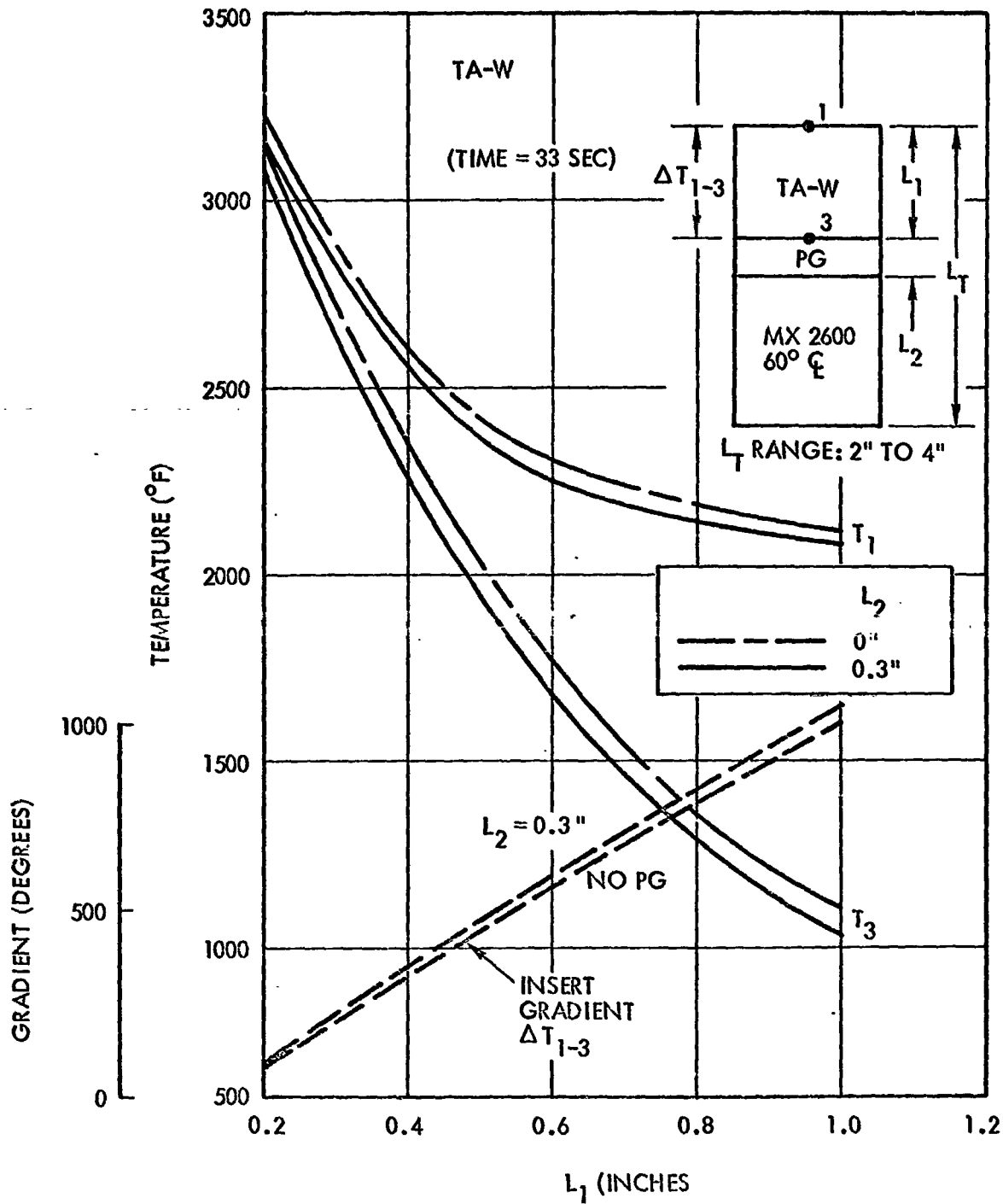


FIGURE 4-24 INITIAL BURN MAXIMUM TEMPERATURE GRADIENT WITH TANTALUM-TUNGSTEN INSERT

- The inclusion of 0.3 inches of Pyrolytic Graphite resulted in a 800°F lower interface temperature at the end of the mission duty cycle and had little or no effect upon back-wall temperatures and insert thermal gradients.
- The insert thermal gradients, interface temperatures and surface operating temperatures were found, in general, to be independent of overall wall thickness and only dependent upon insert material thickness.
- The lower thermal conductivity of the W-Re/ZrO₂ composite and Pyro-Carb inserts resulted in lower interface and backwall temperatures and higher insert thermal gradients than the JTA and Ta-W inserts.

4.1.3 Comparison Study

The selection of a suitable insert material and its dimensions depends upon a trade-off between a wide range of factors. The requirements are:

- A) Maintaining interface temperature below 700⁰F during initial soakback
- B) Maintaining backwall temperature, at the end of the duty cycle, below 800⁰F
- C) Maintaining interface temperature, at the end of the duty cycle below 3000⁰F

The effect of these three requirements upon selection of a suitable insert design will be discussed below for each selected material.

Pyro-Carb

For the maximum insert thickness of 0.75 inches together with an overall wall thickness of 3 inches the maximum backwall temperature is only 380⁰F. For the maximum insert thickness of 1.5 inches studied in this report it is very unlikely that the backwall temperature would exceed 800⁰F. The effect of the insertion of a layer of pyrolytic graphite would result in a slight decrease in backwall temperature.

For the range of dimensions studied, the interface temperature at the end of the duty cycle always exceeds 3000⁰F. Increasing the insert thickness would result in a slight decrease in temperature. However, the results for the other materials indicate that a far more favorable trade-off is obtained by inserting a layer of pyrolytic graphite.

In conclusion, if Pyrocarb is selected as an insert material, all three thermal requirements can be met only if a layer of pyrolytic graphite of at least 0.3 inches is inserted between the insert and the ablator.

JTA Graphite

The results indicate that in order to satisfy requirement B the insert thickness should be no greater than 1 inch for an overall wall thickness of 3 inches. The effect of inserting a layer of pyrolytic graphite is small but favorable.

In order to meet requirement A, Figure 4-7 indicates that for a maximum insert thickness of 1 inch a layer of pyrolytic graphite at least 0.4 inches thick must be used. Referring back to Figure 4-16 indicates that insert dimensions and pyrolytic graphite dimensions of this order would certainly satisfy requirement C.

Wire Reinforced Zirconia

The results indicate that any combination of insert thickness and pyrolytic graphite thickness would satisfy requirements A and B. However, to satisfy the requirements of maximum interface temperature at the end of the duty cycle it is seen that if no pyrolytic graphite is used the insert thickness must be close to 1.5 inches but that insertion of layers of pyrolytic graphite again results in a very favorable trade-off and lower insert thicknesses. For pyrolytic graphite thicknesses of 0.3 inches or greater, any insert thickness satisfying requirement A will satisfy all the other requirements.

Tantalum Tungsten

The results indicate that maximum backwall temperatures would exceed 800°F if insert thicknesses were greater than 1.5 inches. Figure 4-7 indicates that for this thickness a layer of pyrolytic graphite of at least 0.2 inches is required.

Referring back, Figure 4-23 indicates that with these dimensions the interface temperature at the end of the duty cycle would be greater than 3000°F. To meet this latter requirement, an insert thickness of no greater than 1 inch and a pyrolytic graphite thickness of 0.3 inches is required. From Figure 4-7, an insert thickness of 1.0 inch would require a Pyrolytic graphite thickness of at least 0.4 inches.

Conclusions

Following is a summary of the dimensional requirements for all insert materials to satisfy all three thermal requirements, assuming a total wall thickness of 3 inches.

Insert Material	Thickness Requirements
Pyro-Carb	$L_1 = 0.9"$, $L_{PG} = 0.3"$ linearly to $L_1 = 0.43"$, $L_{PG} = 0.6"$
JTA Graphite	$L_1 = 1.0"$, $L_{PG} = 0.4"$ linearly to $L_1 = 0.7"$, $L_{PG} = 0.6"$
Wire Reinforced Zirconia	$L_1 = 1.5"$, $L_{PG} = 0$ $L_1 = 0.6"$, $L_{PG} = 0.3"$ $L_1 = 0.2"$, $L_{PG} = 0.6"$
Tantalum Tungsten	$L_1 = 1.0"$, $L_{PG} = 0.4"$ linearly to $L_1 = 0.67"$, $L_{PG} = 0.6"$

It should be noted that the above configurations satisfy only three of the requirements. Other factors which must be considered in the selection of a suitable insert and its dimensions are thermal gradients, mechanical behavior and thermochemical reactions at the interface.

4.2 FINAL DESIGN CONCEPT

The thermal analysis of the conical LMDE hard throat design has been completed. This study was made in order to evaluate the temperature history of the hard throat insert (Pyrocarb), the backup material (silica phenolic) and the titanium outer case. The entire nominal LMDE duty cycle was used for the imposed thermal environment. The heat transfer test data from the Contour "A" test series was utilized to define the hot side heat flux levels during the duty cycle.

The backwall case temperature was maintained at less than 800°F during the firing in order to minimize thermal stresses. The silica phenolic/Pyrocarb interface was maintained below 700°F during the initial 33 second burn and the following one hour coast (soakback). The silica phenolic/Pyrocarb interface does exceed 3000°F during the final series of burns but this occurs rather later in the duty cycle when the chamber pressure is relatively low.

Previous studies (Section 4.1) had investigated, in a parametric manner, the thermal behavior of several candidate materials for a LMDE hard throat insert. The candidate materials chosen for the final design were a combination of Pyrocarb 751 and Pyrocarb 300 backed by MX2600 ablative. A sketch of this design as applied to the LMDE contour "A" conical chamber is shown in Figure 4-25. An analysis was performed to establish the thermal adequacy of this design and the results are presented in this section.

The three criteria investigated to establish thermal adequacy were as follows:

- a) Maximum backwall temperature throughout duty cycle should not exceed 800°F.
- b) Interface temperature between Pyrocarb and ablative should not exceed 700°F during first firing and subsequent coast. This will prevent charring of the ablative prior to the final duty cycle burn.
- c) Interface temperature between the Pyrocarb and ablative should not exceed 3000°F during complete duty cycle. This is to prevent dimensional ablation of the ablative material. However,

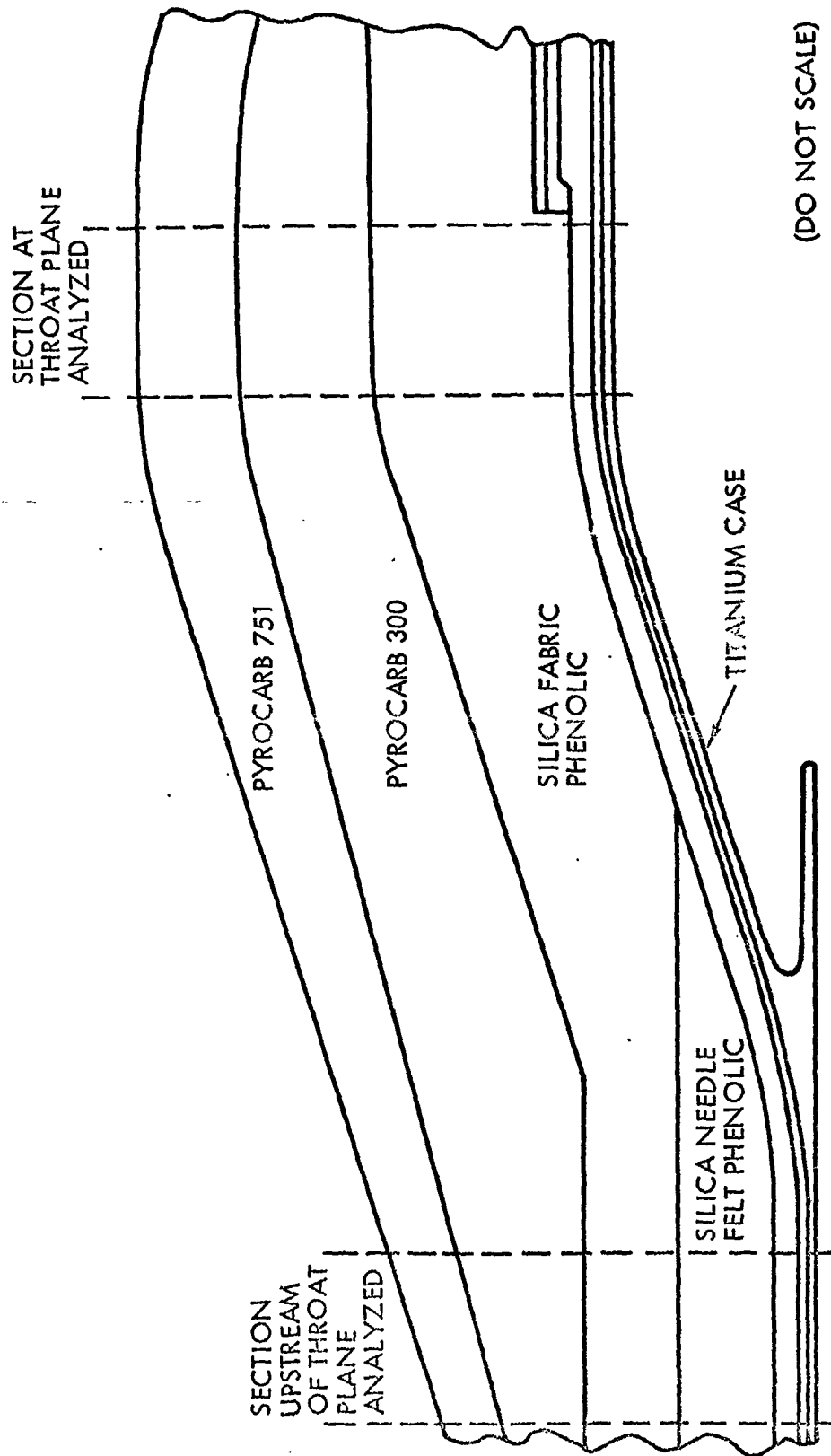


FIGURE 4-25 LMDE CONTOUR "A" CONICAL HARD THROAT DESIGN

time at high temperature is important in order to establish how detrimental the dimensional ablation might be should it occur.

Analysis

One-dimensional thermal models were set up at two different locations on the insert. As indicated in Figure 4-25 one of these locations was at the throat plane and the other was upstream of the throat plane. The thermal operating boundary conditions, gas-side heat transfer coefficients and recovery temperatures are summarized in Table 4-7 for the complete LMDE duty cycle.

The gas recovery temperature at throat plane was derived by using the heat flux data at the throat plane for the Contour "A" chamber and then utilizing the relationship

$$T_r = \frac{(Q/A)}{h} + T_w \quad (1)$$

where Q/A = heat flux to wall at throat plane

h = Bartz heat transfer coefficient

T_w = wall temperature

The heat transfer coefficient at the location upstream of the throat plane was determined by assuming that the recovery temperature was the same as the throat plane and using heat flux data for the upstream location in conjunction with Equation 1 above.

Results and Discussion

The thermal models were formulated for use in conjunction with the TRW Thermal Analyzer Program (TAP) and the results are presented in Figures 4-26 through 4-29.

- Figures 4-26 and 4-27 show surface, interface and backwall temperatures as a function of time at the throat plane.
- Figures 4-28 and 4-29 show surface, interface and backwall temperatures as a function of time for the location upstream of the throat plane.

TABLE 4-7 LMDE DUTY CYCLE FOR CONTOUR A

Time (seconds)	Thrust Level (%)	Heat Transfer Coefficient h_{gas} , (Btu/In ² -Sec-°F)		Recovery Temperature (°F)
		Throat Plane	Upstream of Throat Plane	
<u>Initial Burn</u>				
0 - 25.999	10	0.145×10^{-3}	0.133×10^{-3}	3650
26 - 32.999	100	0.72×10^{-3}	0.515×10^{-3}	3560
<u>Coast</u>				
33 - 3632.999	0	1.0×10^{-12}	1.0×10^{-12}	0
<u>Final Burn</u>				
3633 - 3658.999	10	0.145×10^{-3}	0.133×10^{-3}	3650
3659 - 4022.999	100	0.72×10^{-3}	0.515×10^{-3}	3560
4023 - 4098	60	0.48×10^{-3}	0.395×10^{-3}	3950
4223	40	0.37×10^{-3}	0.318×10^{-3}	3570
4233	30	0.28×10^{-3}	0.270×10^{-3}	3610
4533	20	0.19×10^{-3}	0.180×10^{-3}	3630
4600	0	1.0×10^{-12}	1.0×10^{-12}	0

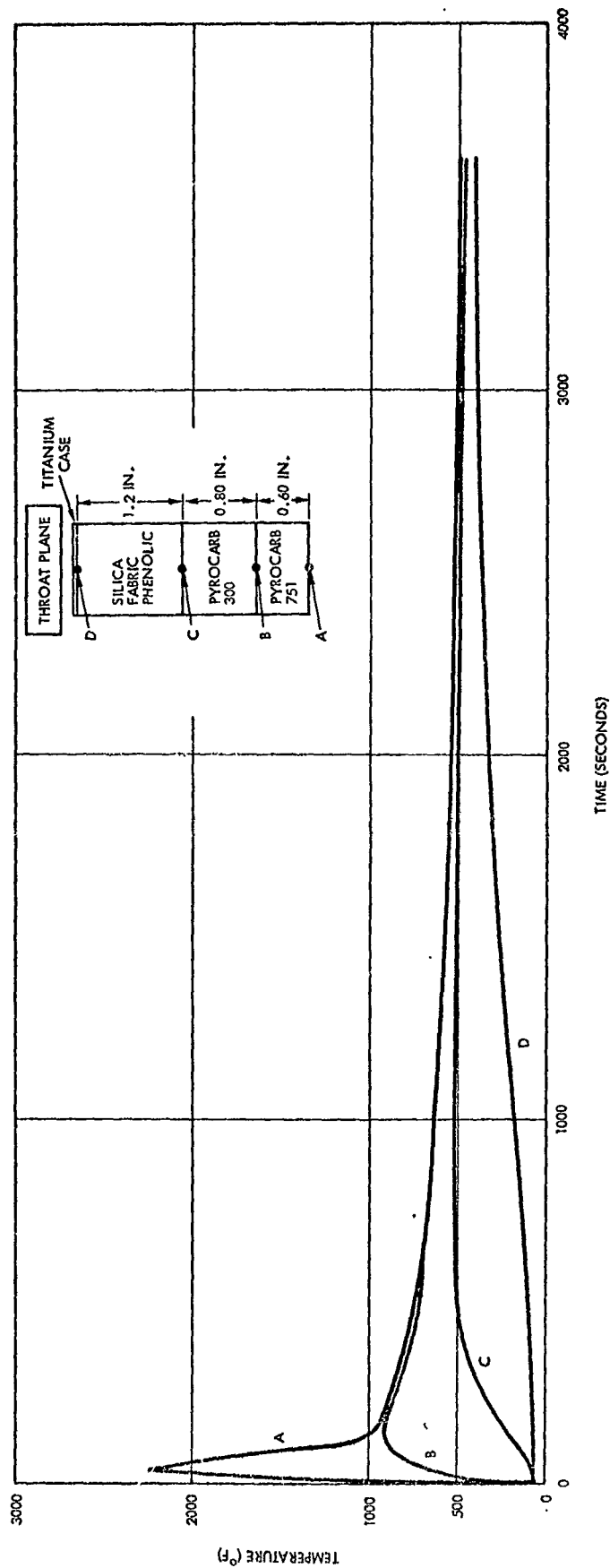


FIGURE 4-26 HARD THROAT TEMPERATURE HISTORY DURING
INITIAL BURN AND SOAKBACK - THROAT PLANE

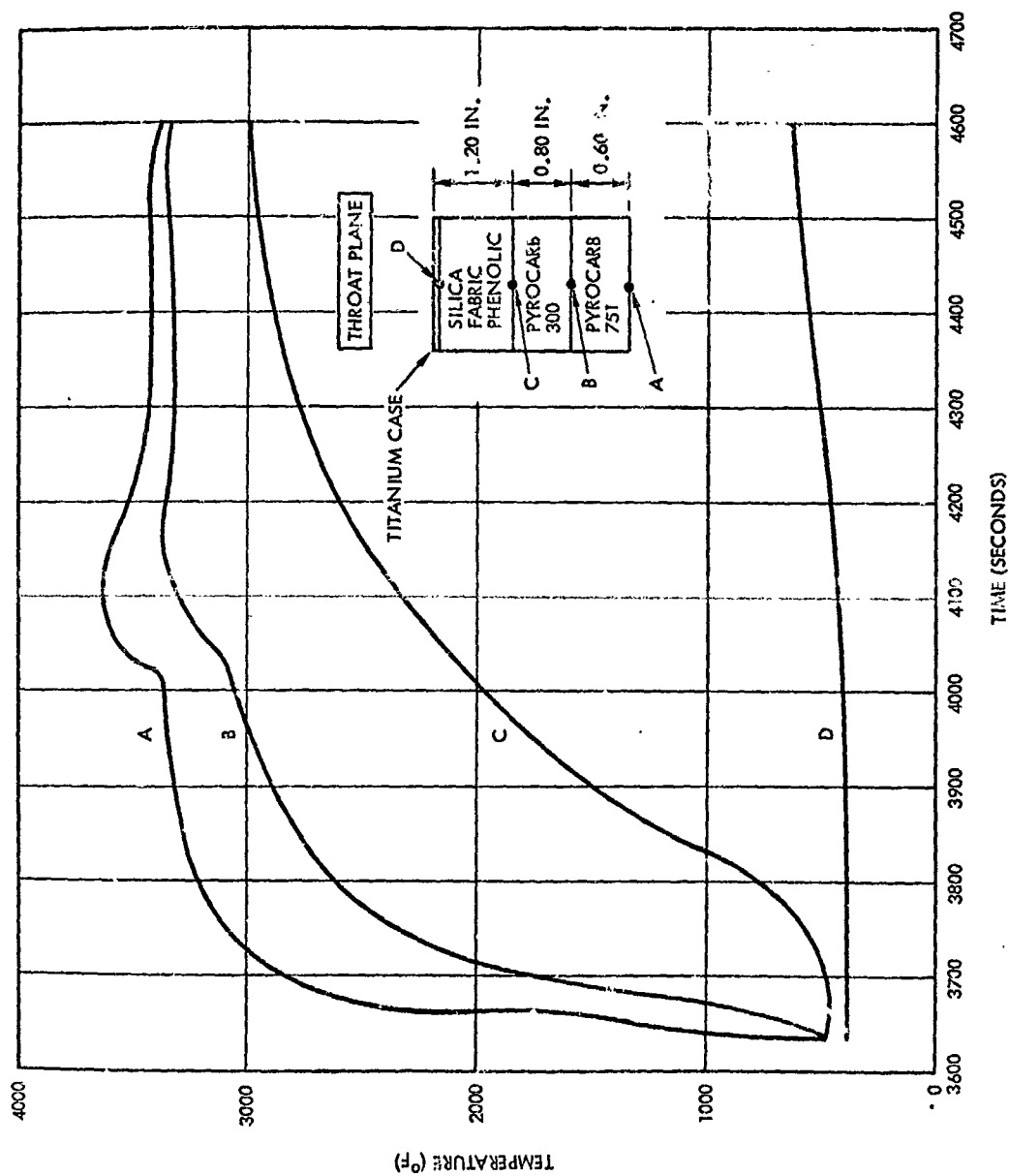


FIGURE 4-27 HARD THROAT TEMPERATURE HISTORY
DURING FINAL BURN - THROAT PLANE

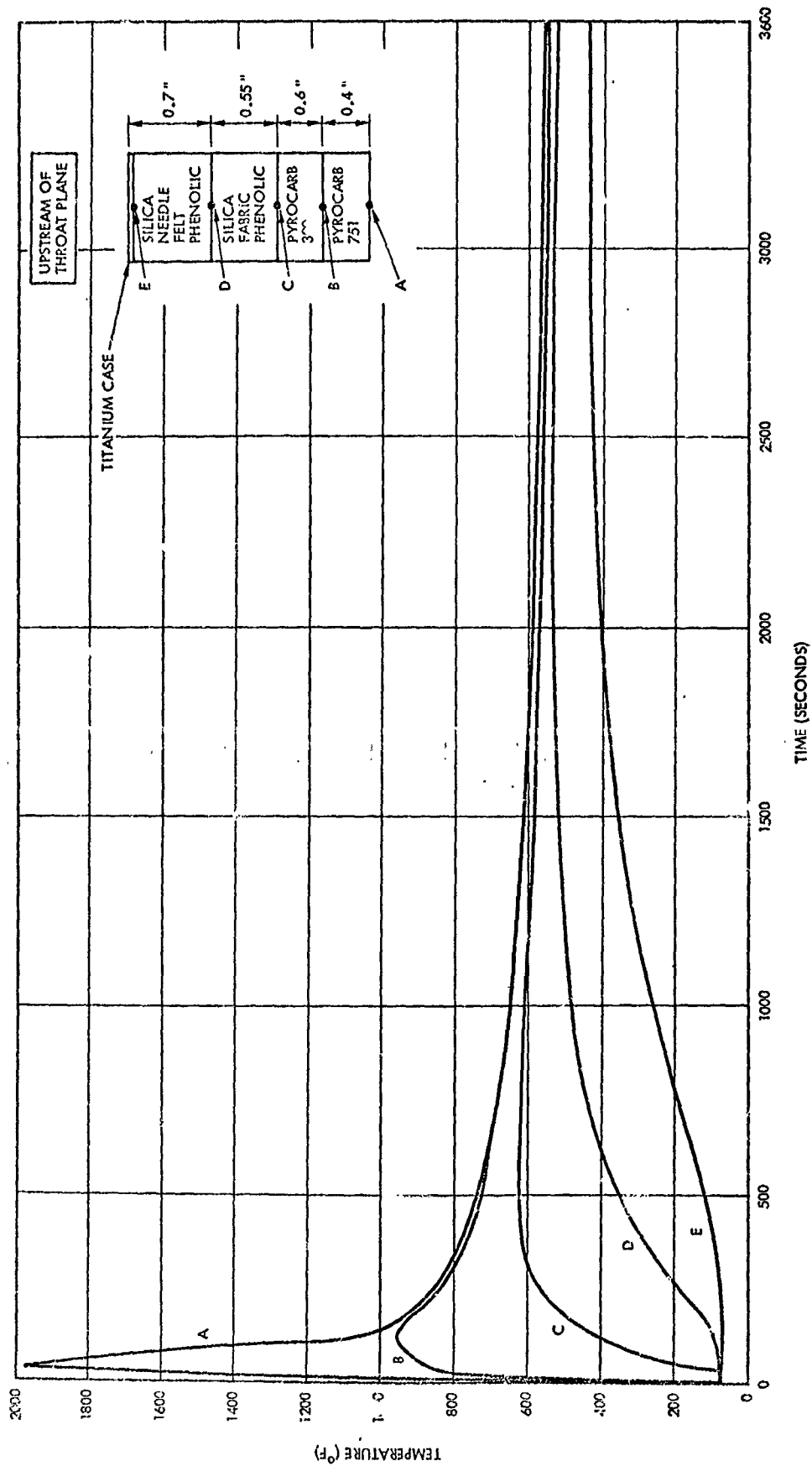


FIGURE 4-28 HARD THROAT TEMPERATURE HISTORY DURING INITIAL BURN AND SOAKBACK -- UPSTREAM OF THROAT PLANE

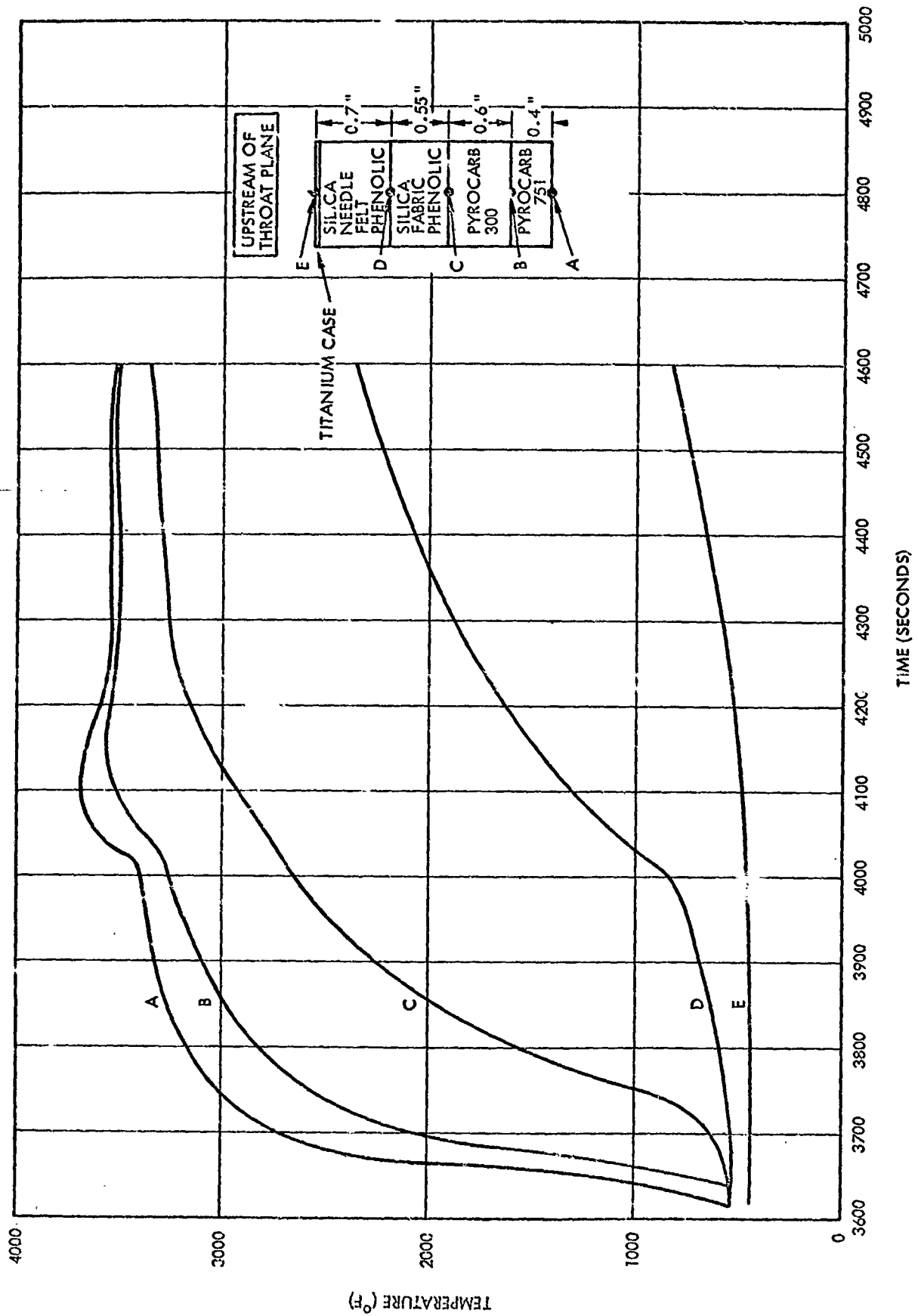


FIGURE 4-29 HARD THROAT TEMPERATURE HISTORY DURING
FINAL BURN - UPSTREAM OF THROAT PLANE

These results reveal the following:

- The backwall (titanium case) temperature remains below 800°F throughout the duty cycle at both locations.
- The interface temperature between the Pyrocarb and the ablative remained well below 700°F during the first firing and the subsequent coast.
- During the final series of burns the interface between the Pyrocarb and ablative does exceed 3000°F. In the case of the throat plane this doesn't occur until 4580 seconds into the duty cycle (end of burn at 4600 seconds) when the chamber pressure is less than 20%. In the case of the location upstream of the throat plane this occurs at 4130 seconds into the duty cycle when the thrust level is 50%.

5.0 ABLATIVE CHAMBER REACTIVE TESTS

Four full-ablative chambers were test fired as part of the CIC Program. Results of a fifth test are included in this report because of its pertinence to the program.

Three of the tests were conducted with sea-level ($\epsilon = 2$) ablative chambers to measure throat erosion during the mission duty-cycle with LM Head End Assembly, S/N 023, assigned to this program. They include:

- A baseline Qualification B configuration chamber to establish duty cycle erosion characteristics of HEA S/N 023.
- A contour A chamber, the configuration that showed the lowest heat flux at full thrust in heat sink tests.
- A four-quadrant composite throat to compare erosion characteristics of three advanced materials to the standard MX-2600 material. The configuration for this test was Contour C, which exhibited the best c^* performance in the heat sink and water-cooled chamber tests.

The final two tests were conducted in the high altitude test facility with flight configuration chambers. A mainline LMDE head end assembly, S/N 034, was assigned so that a direct performance comparison could be made with the Qual B LMDE chamber for an extended G mission duty-cycle planned for Apollo 16 and subsequent. The tests included:

- An advanced prototype chamber, in the Contour C configuration, that evolved from this program and contained a quartz-fabric/phenolic turbulence ring and throat.
- A Qual B configuration chamber that also incorporated quartz-fabric/phenolic in the turbulence ring and throat.

5.1 BASELINE QUALIFICATION B CHAMBER

To establish the specific duty cycle erosion characteristics of the LMDE Qualification B head end assembly, S/N 023, assigned to this program a baseline test series was conducted in the Vertical Engine Test Stand

(VETS) with a 2:1 expansion ratio ablative chamber. Complete results are reported in Reference 16. The test series is summarized in Table 5-1.

TABLE 5-1. Test Sequence

<u>Test Description</u>	<u>Test Designation</u>	<u>Test Duration (Sec.)</u>
Checkout Test	VA1-666	26
Checkout Test	VA1-667	51
Checkout Test	VA1-668	22
Checkout Test	VA1-669	51
Duty Cycle (Phase I)	VA1-670	42
Duty Cycle (Phase II)	VA1-671	<u>822</u>
Total		1,014 seconds

The duty cycle portion of the test was conducted as indicated in Table 5-2, with propellants saturated at 65 psia with gaseous nitrogen.

TABLE 5-2. Baseline Duty Cycle

<u>Test Description</u>	<u>Thrust %</u>	<u>Duration (Hrs.)</u>	<u>Duration (Sec.)</u>
Duty Cycle (Phase I)	15		26
	FTP		7
	25		6
Soak Period		1	
Duty Cycle (Phase II)	15		26
	FTP		320
	60		50
	65		48
	50		66
	40		30
	25		46
	15		25
	40		44
	25		162

- A 25 percent thrust shutdown was added to the first phase of this duty cycle to minimize the possibility of spiking occurring and damaging the injector element.

- The full thrust burn during the second phase of the duty cycle was reduced from 370 seconds to 320 seconds, because of the propellant tank limitations of VETS "A" side and the complexity of supplementing these tanks.

Throat erosion for the duty cycle was fairly uniform, as may be noted in Figure 5-1, with a measured throat area increase of 23.7 percent.

This value compares well with the nominal duty-cycle erosion of Qual B Engines 4, 5, 6 and 7 (Table 1-1), considering the fact that the latter were not exposed to the highly erosive 65% FTP setting, and the baseline test was considered to be representative.

As may be noted from the engine performance data of Figure 5-2, the throat area increase at FTP was 7.6 percent, measured from the time of initial throat swelling. This value compares favorably with the 7.3 percent erosion predicted from the 79 second " T_{120} " time of the fiberglass streak throat test (VA1-659).

Thrust is not measured on the VETS "A" facility. Therefore, an analysis method used early in the LMDE program was updated and used in order to present overall engine performance data as shown in Figure 5-2. This method utilizes a model for c^* (characteristic chamber exhaust velocity) that is a function only of actual measurable quantities of the chamber and injector. The c^* model utilized was of the form produced by the Engine Simulation Group for the purposes of LMDE flight analysis.

The c^* model utilized was adjusted to closely match the pre-duty cycle test data for this engine. Once the c^* is known independently, it is possible to solve for the throat area at any time during the duty cycle.

The only problem encountered in the performance analysis of these tests was caused by the test facility. The water-cooled slip-ring, necessary to attach hard throat chambers to the diffuser, tends to collect the congealed molten silica removed from the chamber and the throat. The buildup remaining in the slip joint at the end of the test is shown in Figure 5-3. When throttling down to approximately 50 percent thrust,

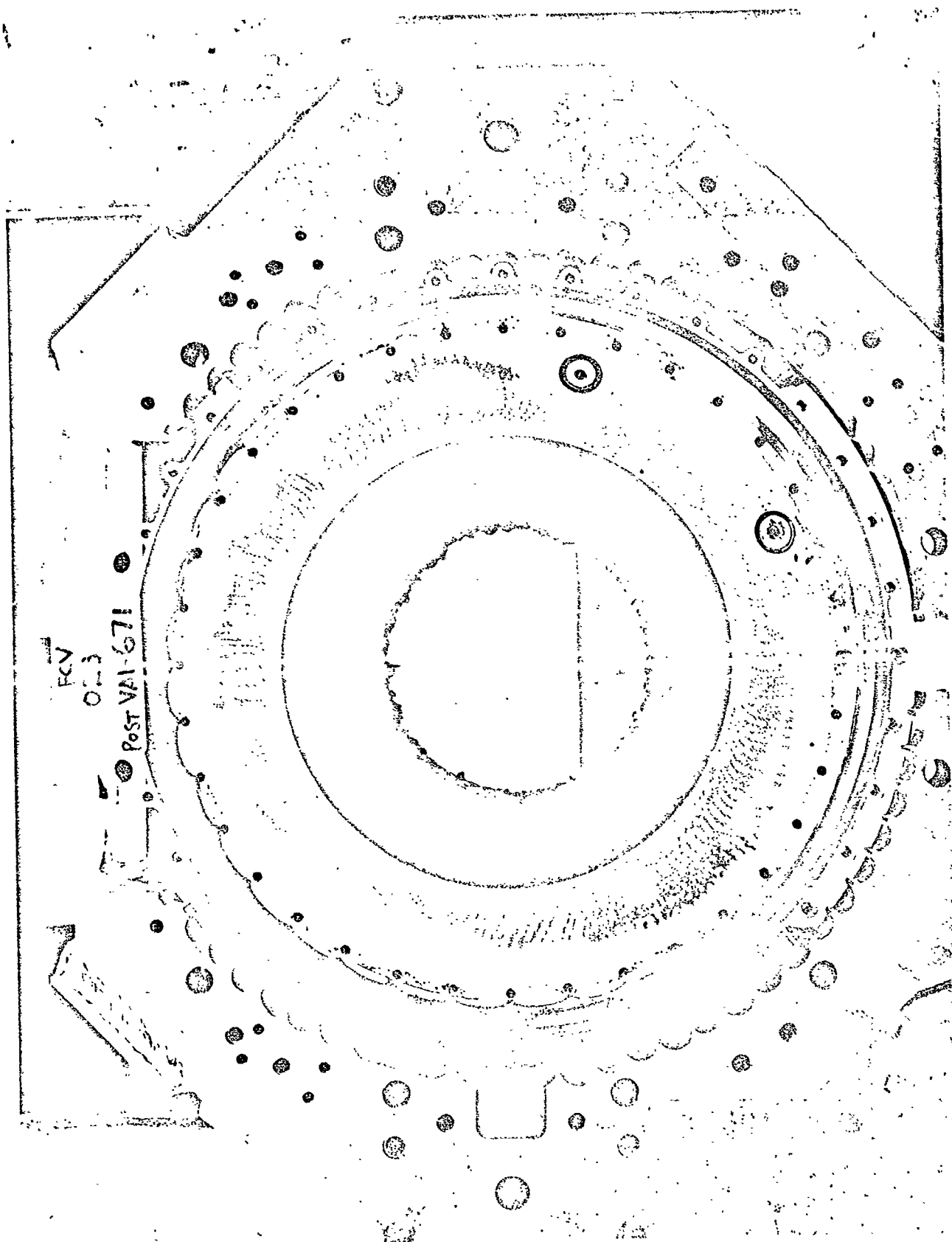
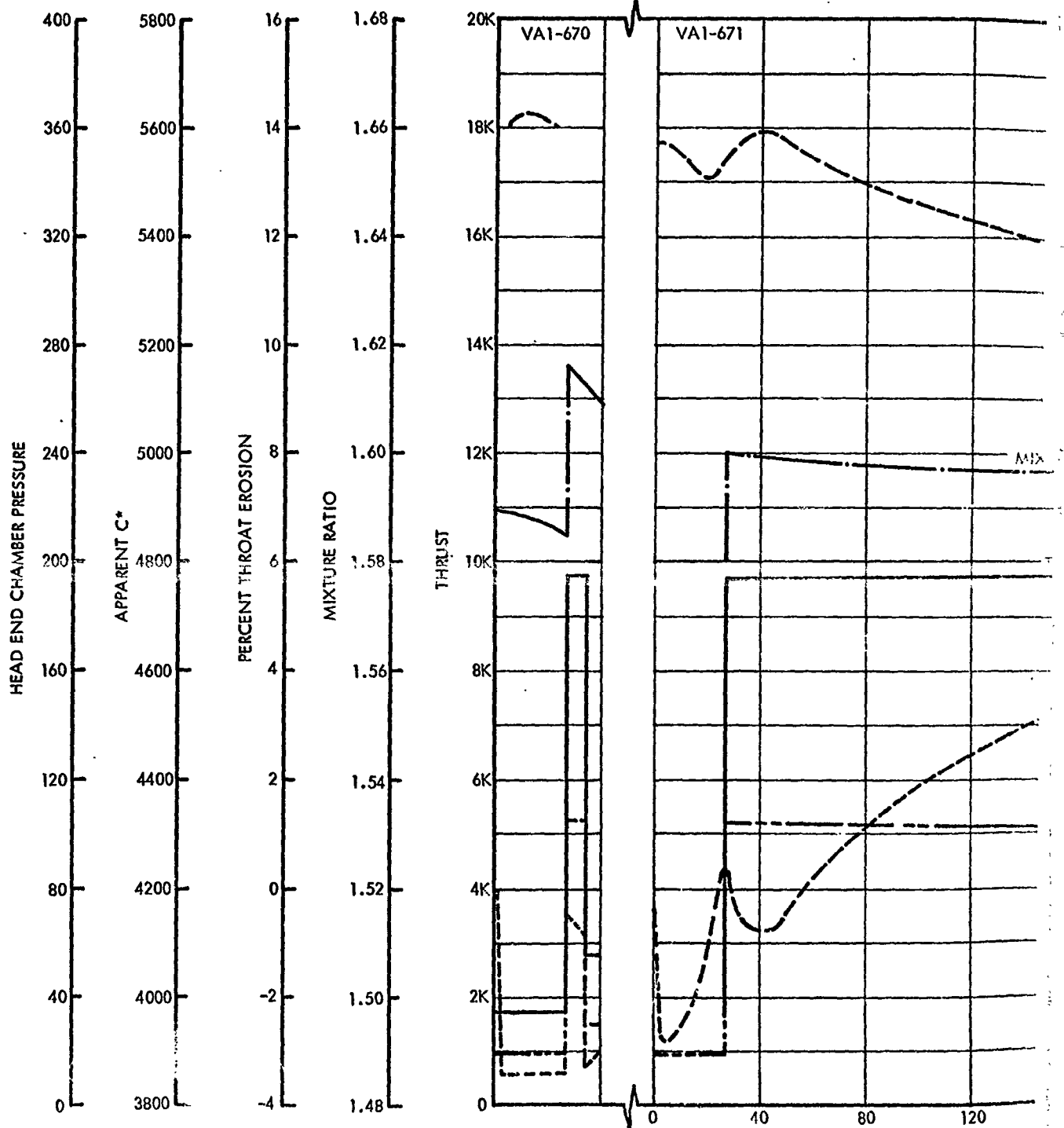
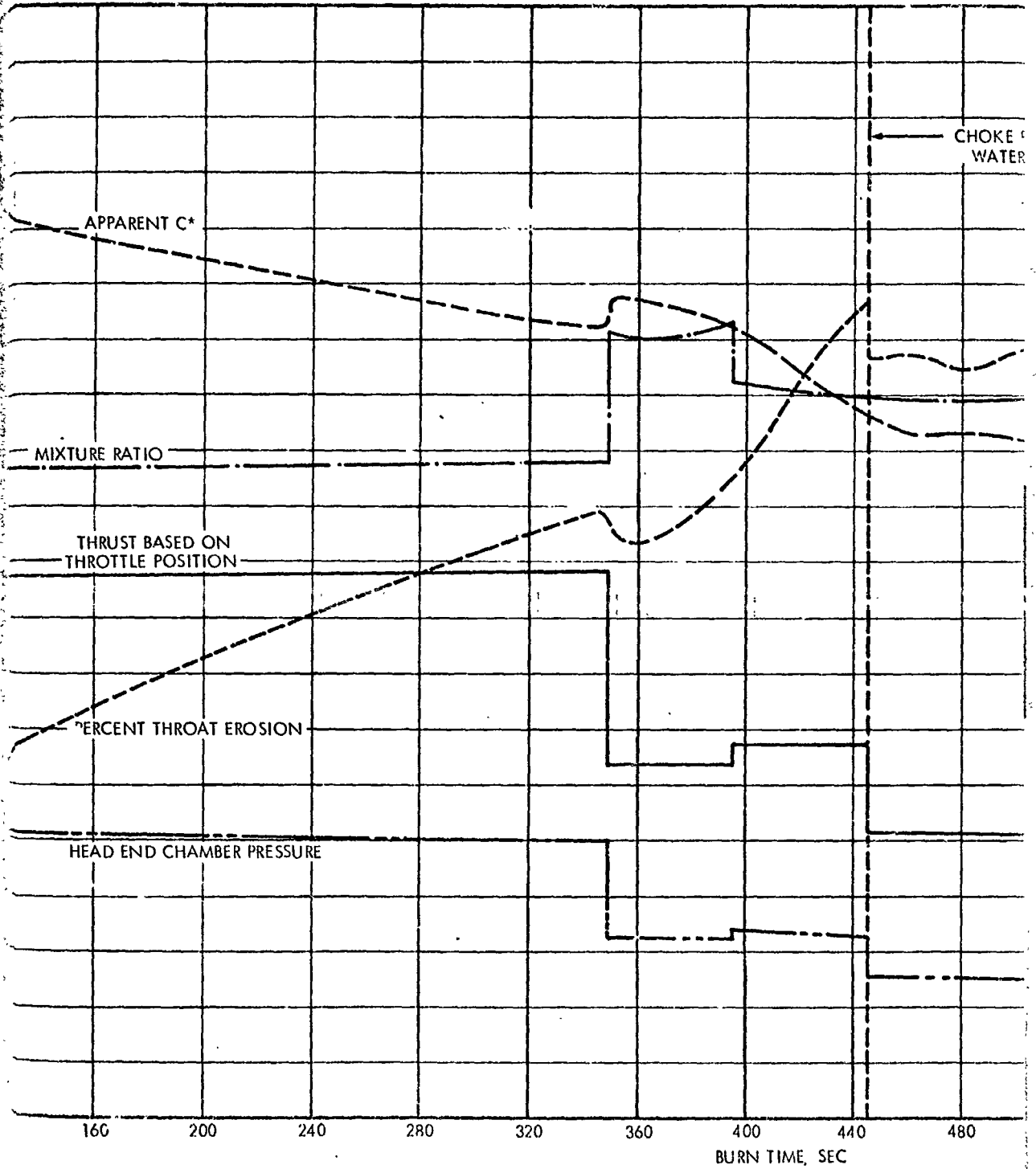


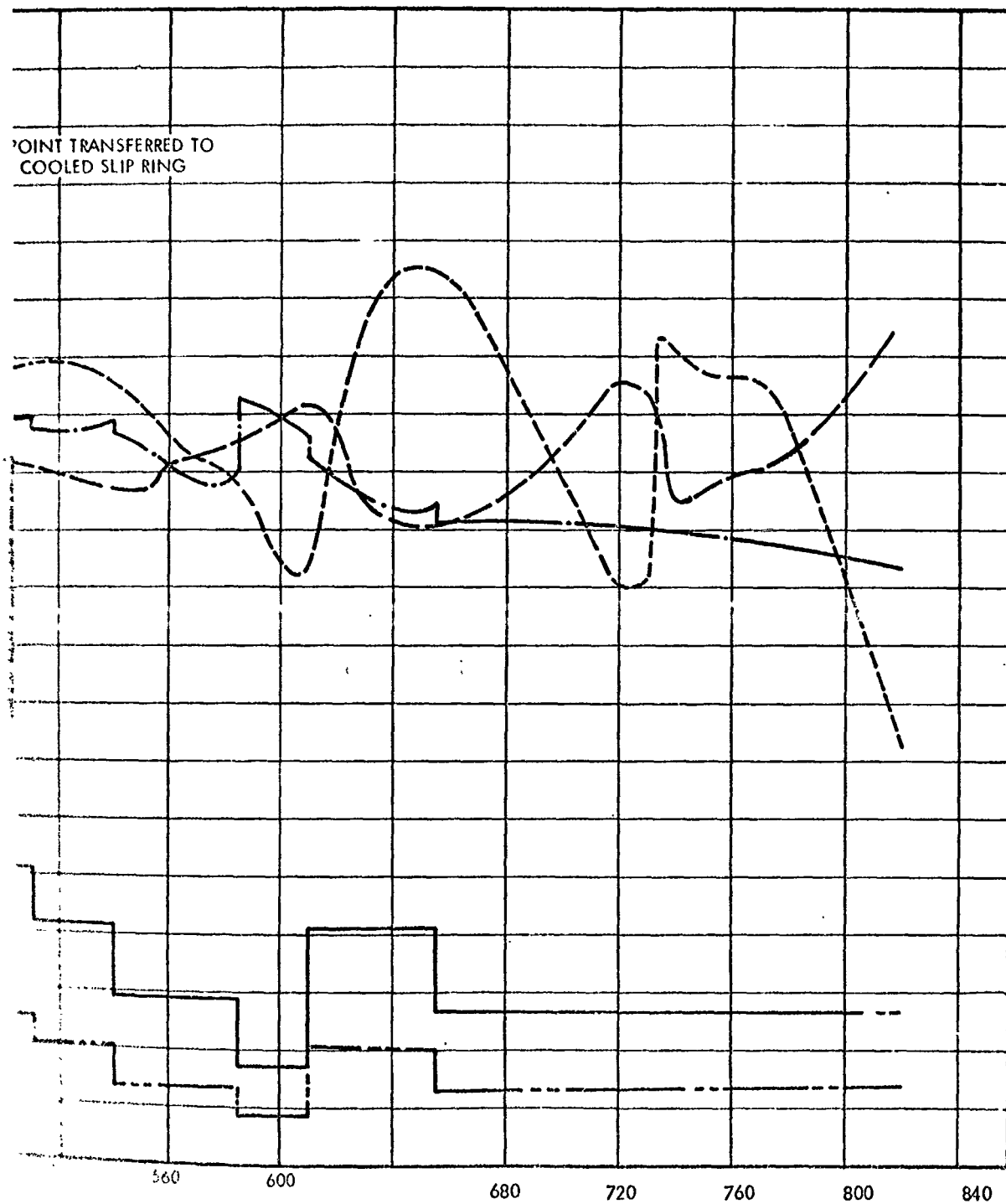
Figure 5-1. Baseline Throat Erosion



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HEA 023 VA1-670 and 671 CTS DATA





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Figure 5-2. Engine Performance Data
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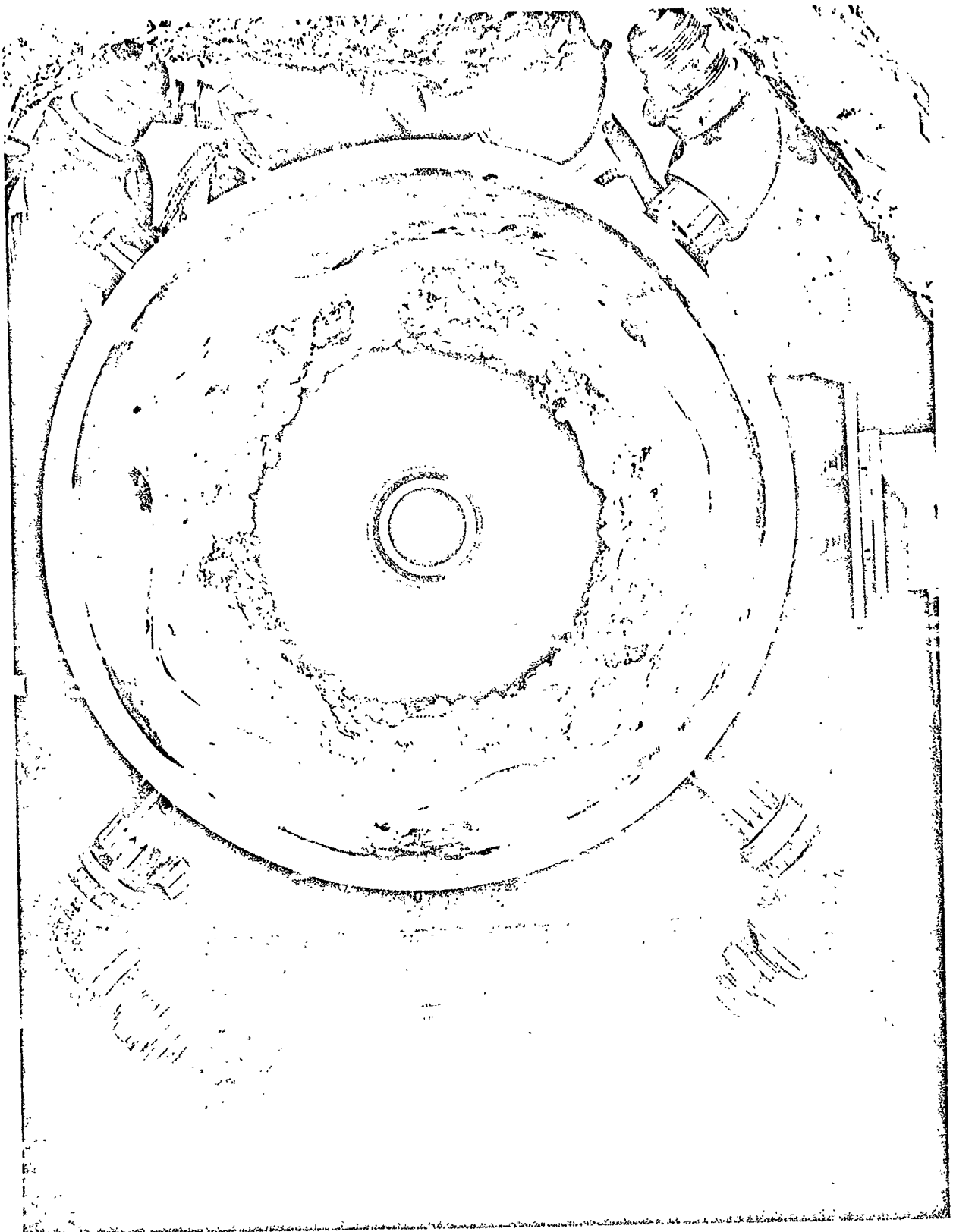


Figure 5-3. Silica Build-up

at 445 seconds into the second phase of the duty cycle, the silica buildup in the slip joint became the controlling station for choked flow. Because the throttle position was always lower than 50 percent for the remainder of the test, this secondary throat remained the choke point for the duration of the test, invalidating all subsequent data with the exception of flowrates. With respect to the internal performance parameter analysis, this meant that the model was solving for parameters not characteristic of a standard LMDE combustion chamber configuration.

Figure 5-2 also indicates the time at which the choke point was transferred to the water cooled slip-ring (see vertical dashed line). The data to the right of this point represent a nonstandard LMDE configuration. It is interesting to note that the throat erosion trace indicates the throat to be expanding and contracting. This is probably due to, first, a buildup and second, erosion of the throat as the deposits on the water-cooled slip-ring buildup and then erode away.

If the full thrust erosion rate at 320 seconds, 7.6 percent, is projected to the normal full thrust burn time of 370 seconds, the erosion at FTP would have been slightly greater than 8.5 percent.

Post-firing condition of the ablative throat, Figure 5-4, shows silica globules that were chipped away so that the post firing throat measurement could be made. These measurements, taken in five degree increments, are plotted in Figure 5-5 and are compared with the erosion profiles of fiberglass streaks fired at FTP and 25 percent thrust. When viewed from the exit end of the chamber with the FCV at 0° , maximum throat erosion was observed at 130° and minimum erosion at 235° . Figure 5-6 shows the ablative liner cross sections at the points of minimum and maximum erosion. As usual in a duty cycle firing, the cylindrical portion of the ablative liner was charred completely through. Zero char was observed in relatively cool regions of the faceplate and in the area of the large diameter cylindrical section near the faceplate. Shrinkage of the light weight back-up material in the cylindrical section and consequent failure in the overwrap probably occurred during heat soakback. The fairly uniform erosion is further demonstrated by the $3/4$ inch to $7/8$ inch virgin material remaining at the throat.

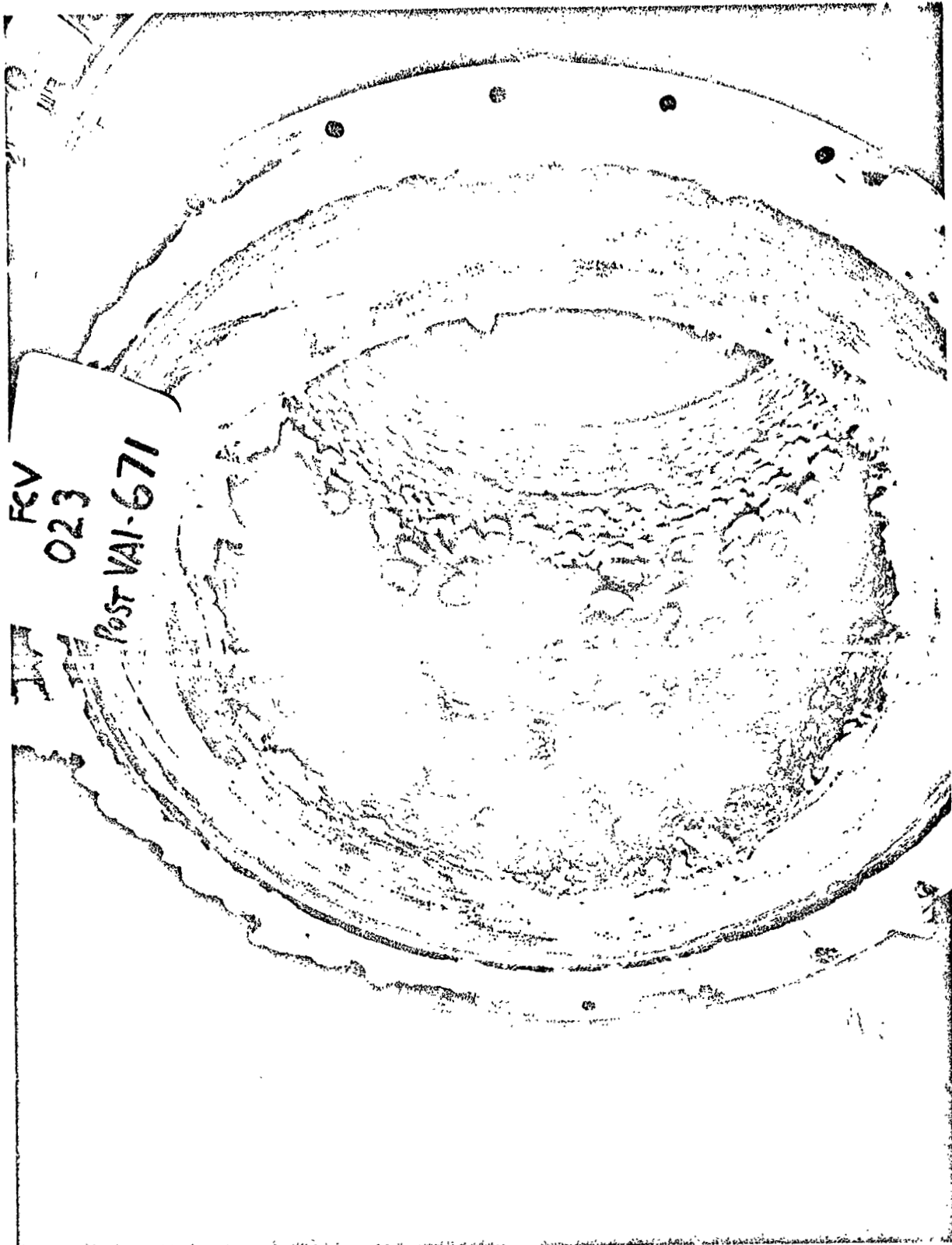
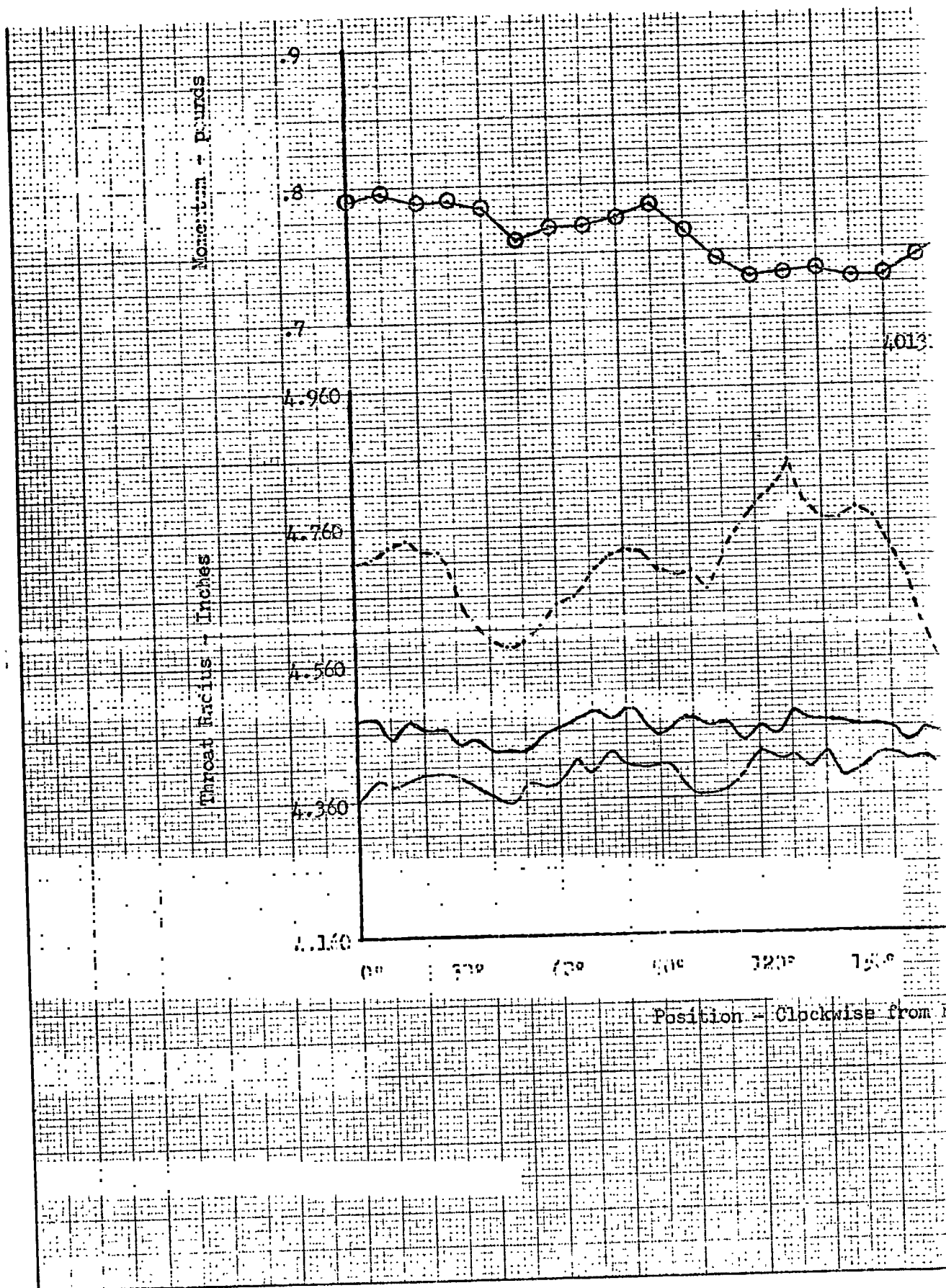


Figure 5-4. Post-Firing Throat Condition



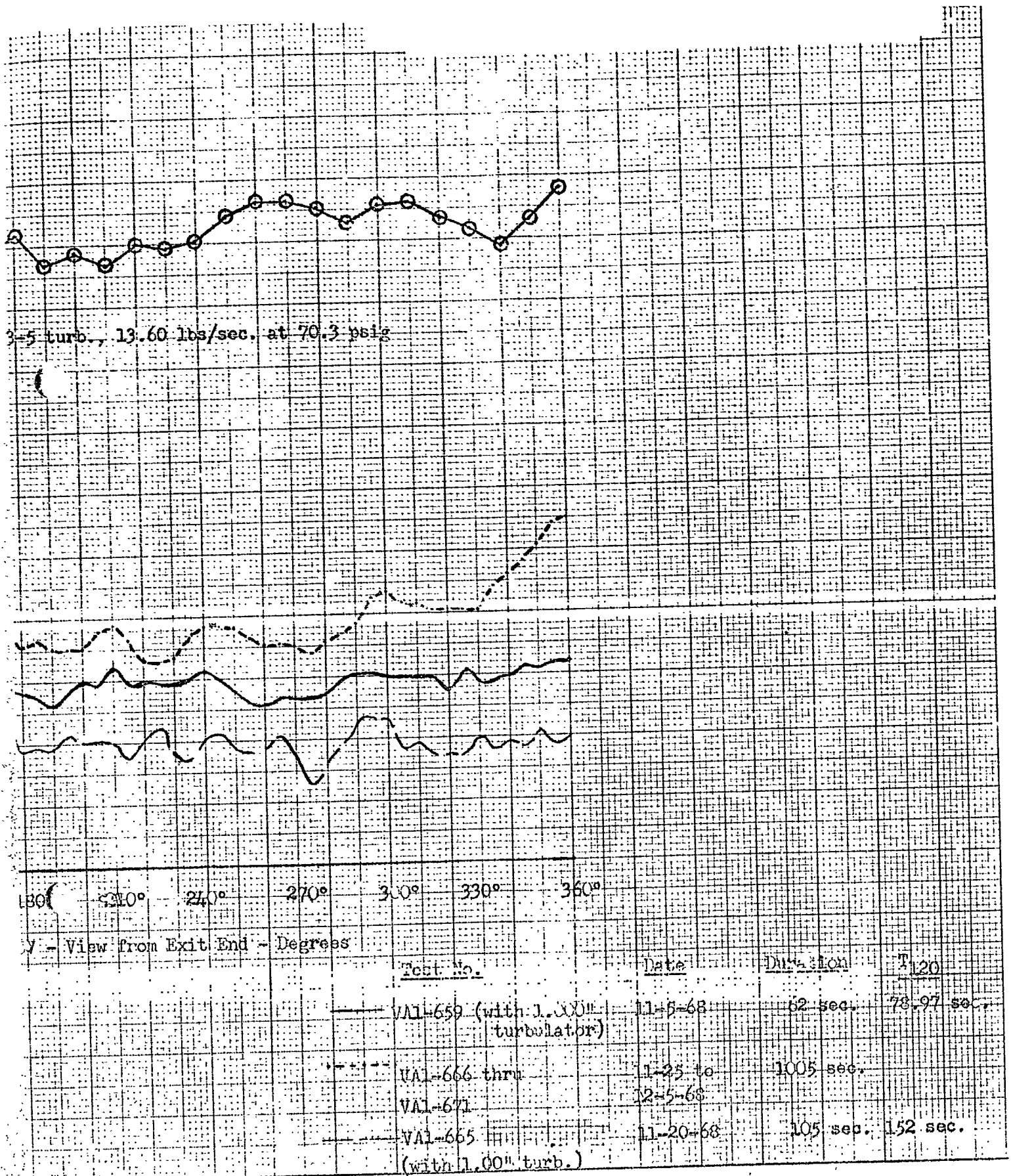


Figure 5-5. Qual B Baseline Throat Erosion Profile

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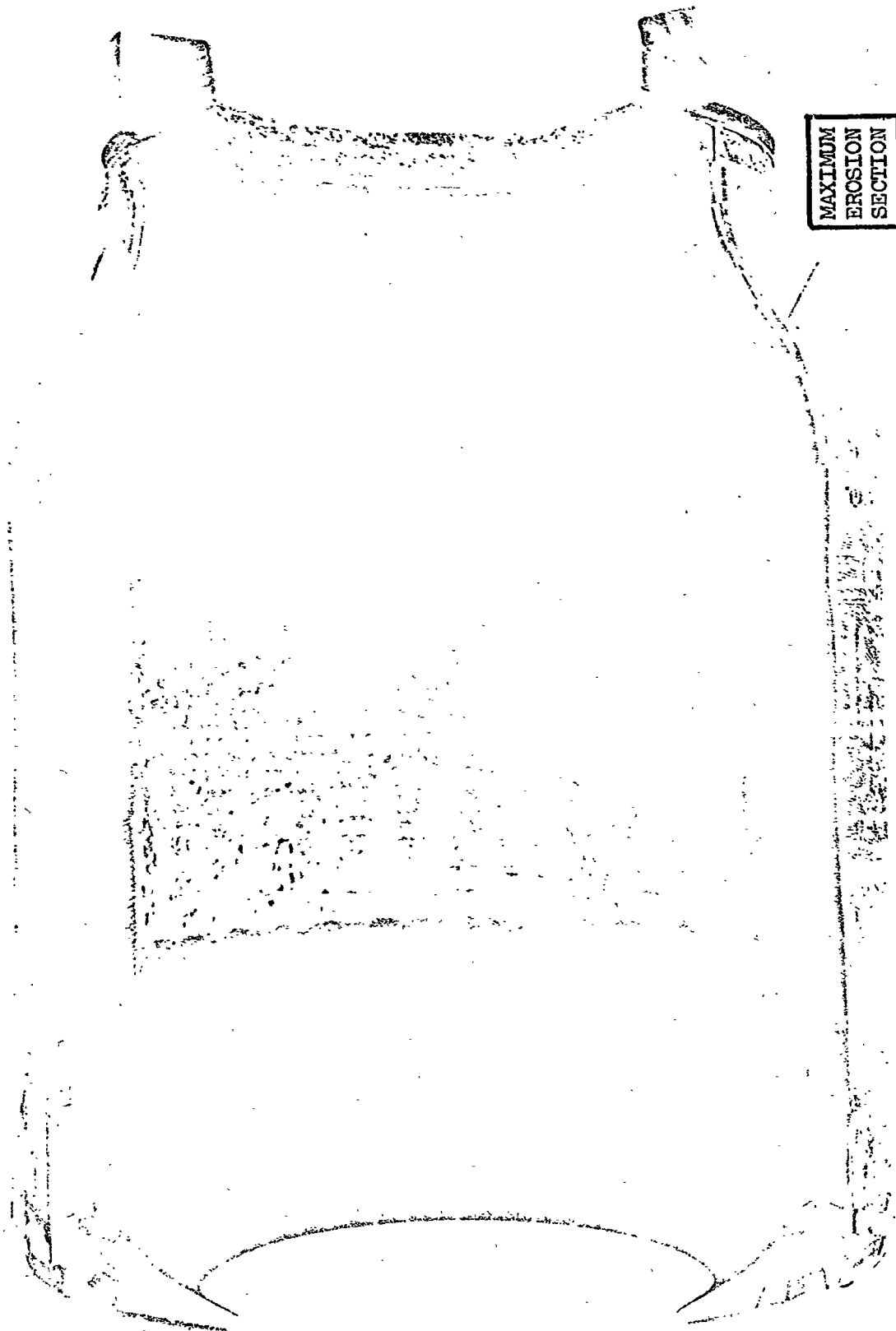


Figure 5-6. Post-Test Chamber Erosion

In Figures 5-7 through 5-10, backwall temperatures measured at various axial stations are further indications of injector and erosion uniformity based on the narrow band of temperatures around the chamber circumference. If the shutdown temperature data at 818 seconds are projected to the nominal duty cycle time of 865 seconds, the temperature range at the various stations would be as shown in Table 5-3. Note that all temperatures are well within the full-thrust design allowable values.

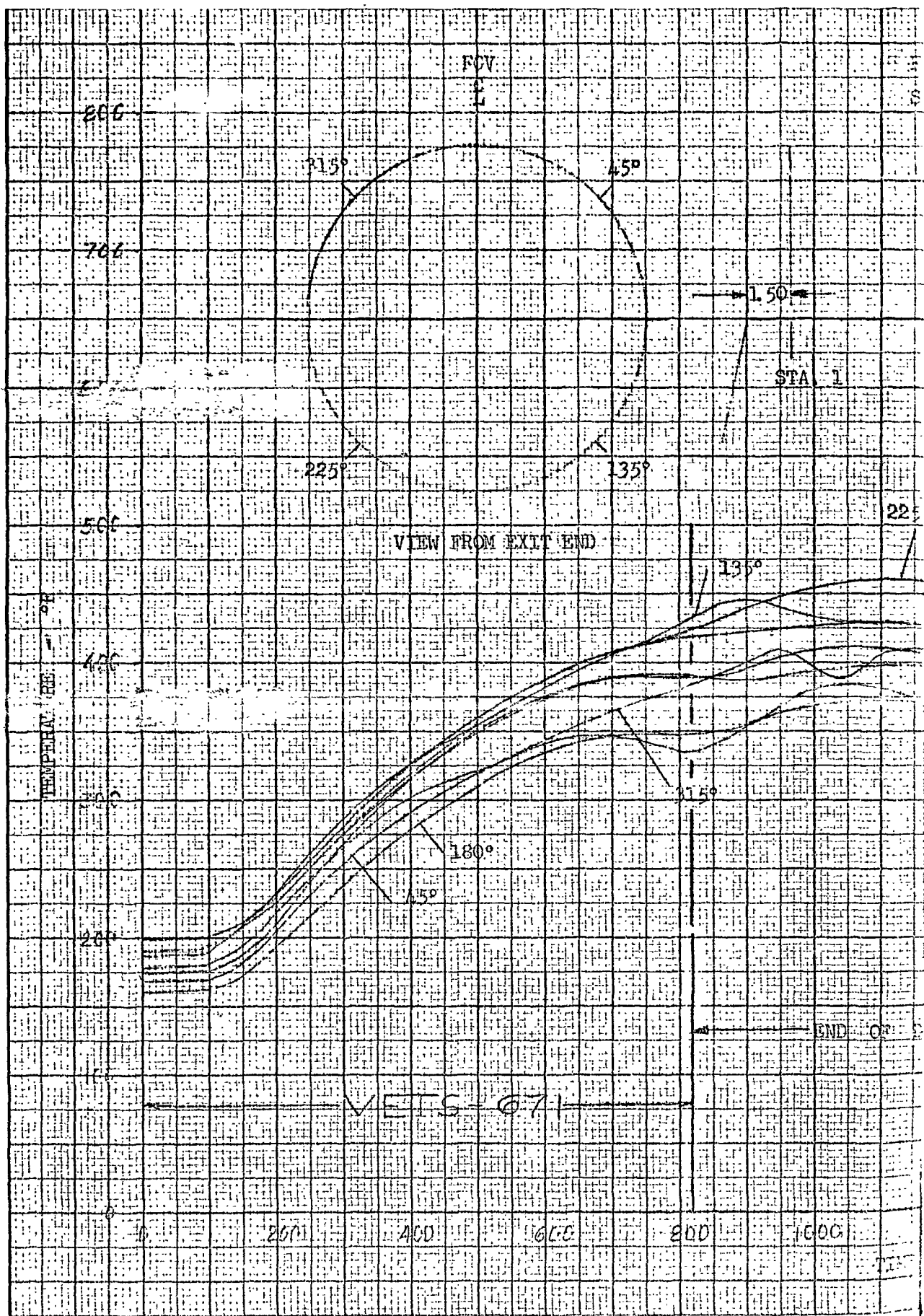
TABLE 5-3. Projection of VA1-671 Data to 865 Seconds

<u>Axial Distance from Injector Attach Flange (inches)</u>	<u>Backwall Temperature (°F)</u>	<u>Full-Thrust Design Allowable Temperature (°F)</u>
1.5	345 - 445	820
11.5	480 - 570	820
16.5	280 - 335	1,005
19.69	285 - 308	1,050

5.2 CONTOUR A TEST CHAMBER

Results from the chamber wall environment tests (Section 3) showed that the lowest heat flux at FTP for the contours evaluated was exhibited by Contour A. Consequently, a sea-level ablative chamber was fabricated to a modified Contour A configuration with the standard LMDE MX-2600 silica-fabric/phenolic. When test fired (VA1-741 through VA1-743), the measured total throat erosion was 28.2 percent for the following nominal baseline duty cycle:

<u>Test Description</u>	<u>Thrust (%)</u>	<u>Duration (Sec)</u>
Duty Cycle	15	26
	FTP	7
	25	6
Soak Period		(1 Hour)
Duty Cycle	15	26
(Phase II)	FTP	320
	60	50
	65	48



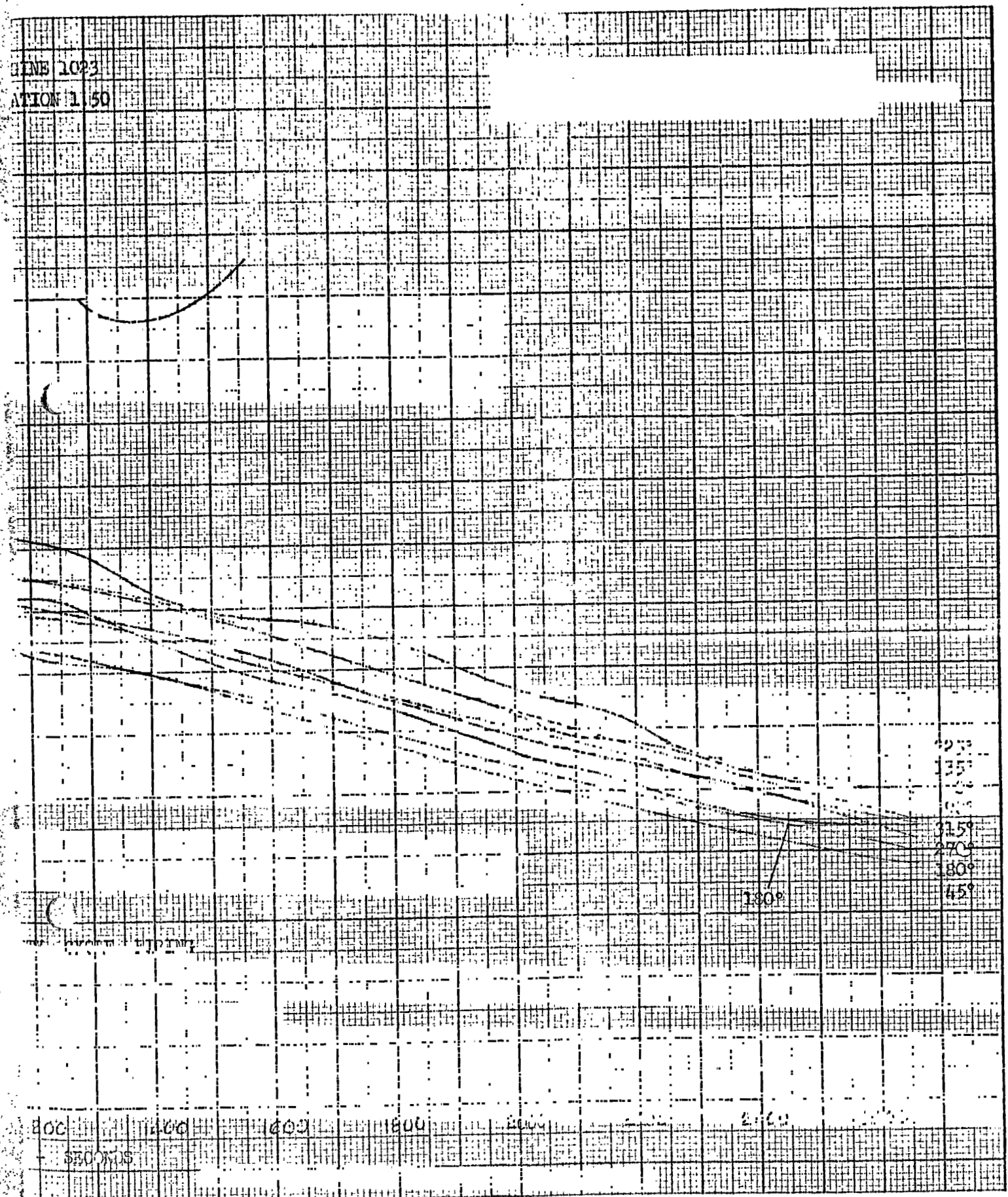


Figure 5-7. Titanium Case Temperature Data

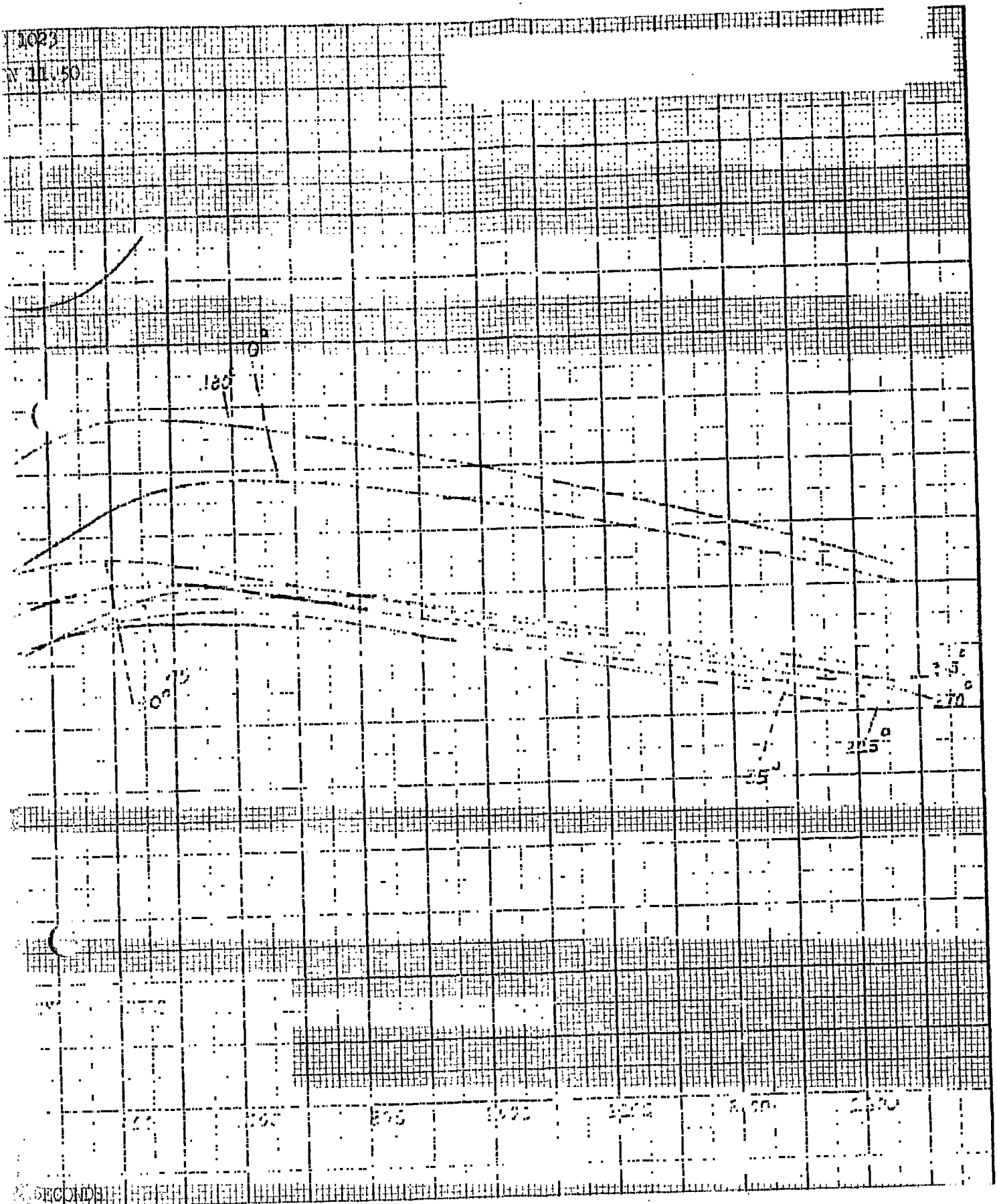
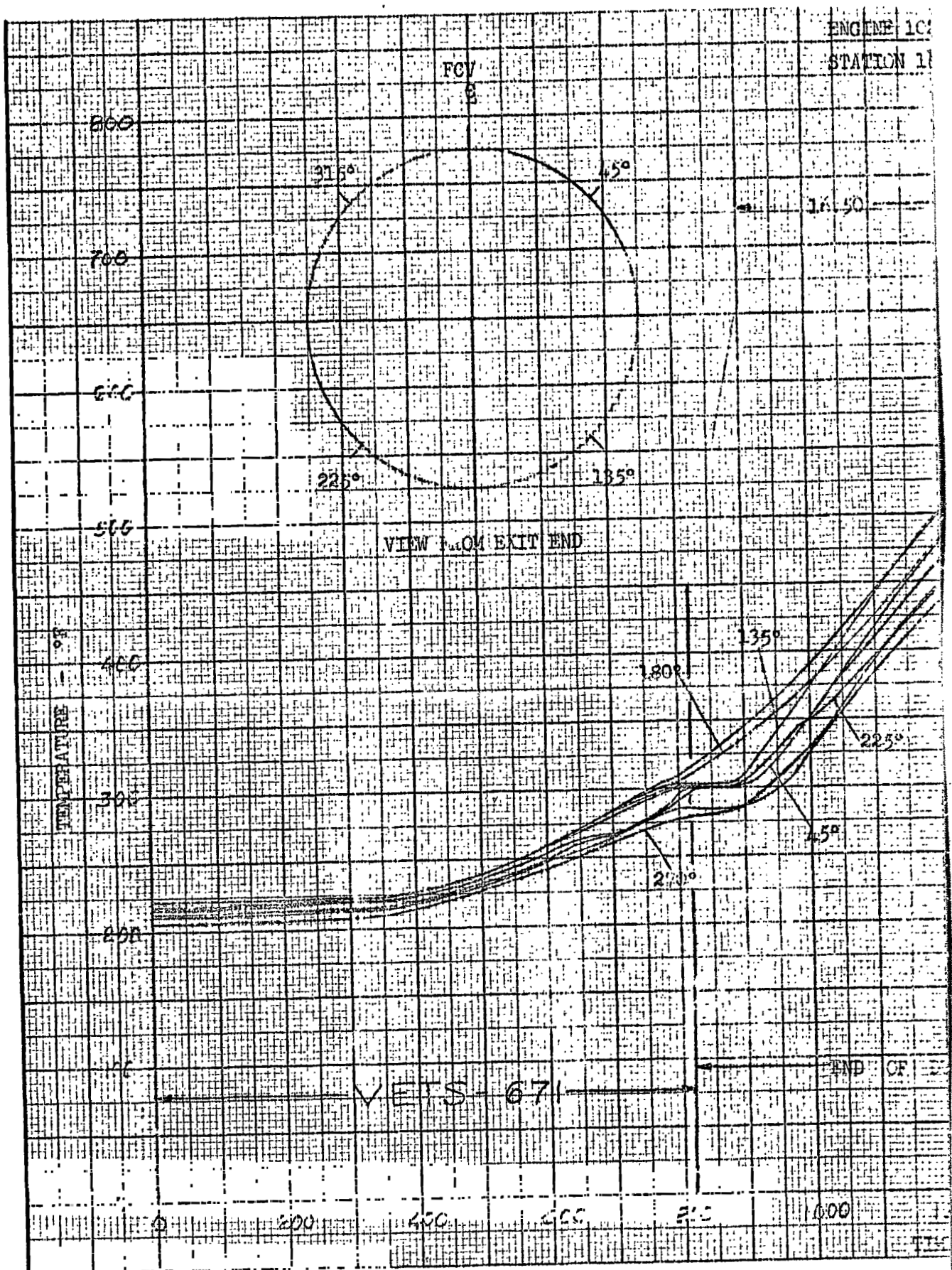


Figure 5-8. Titanium Case Temperature Data



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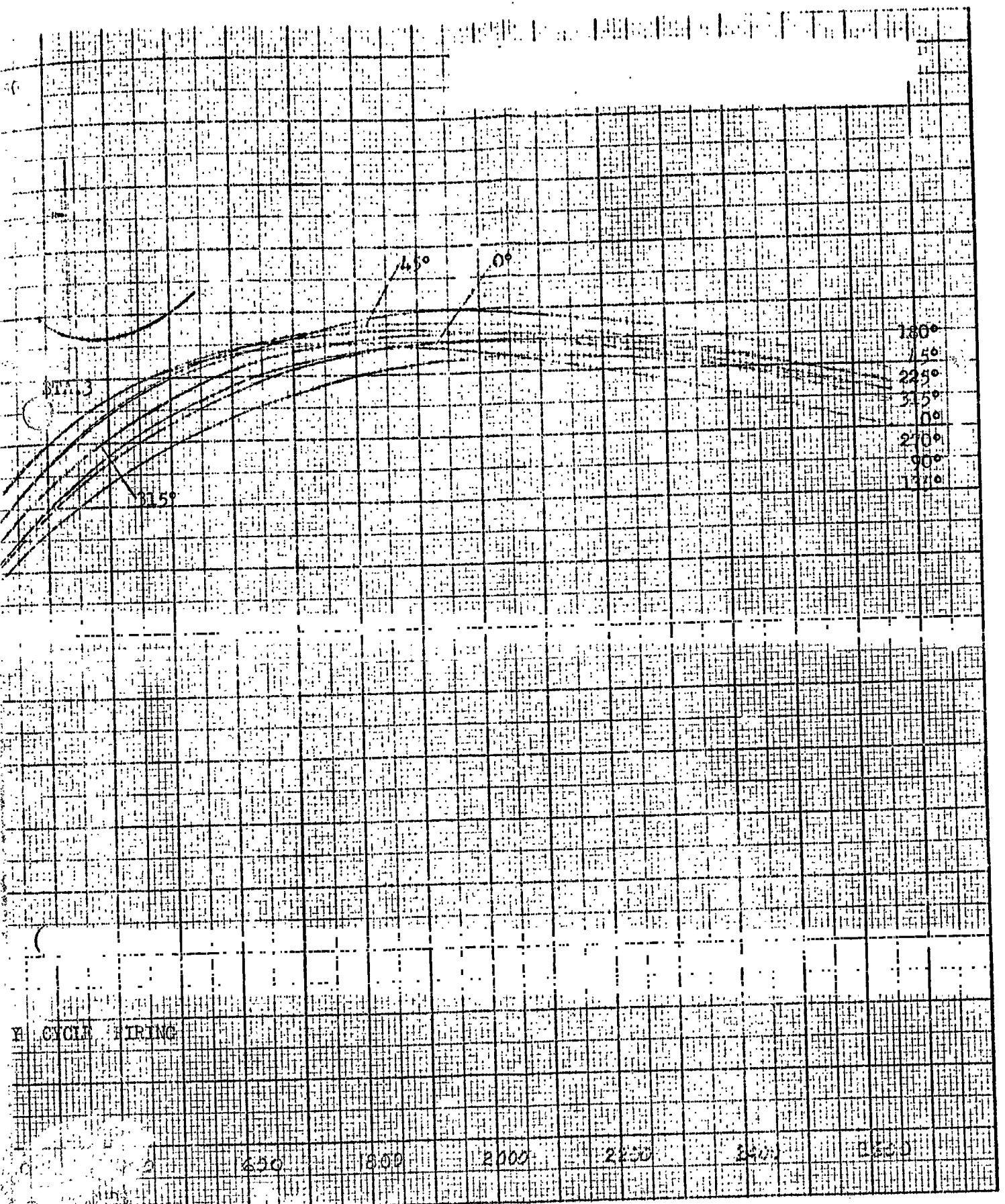
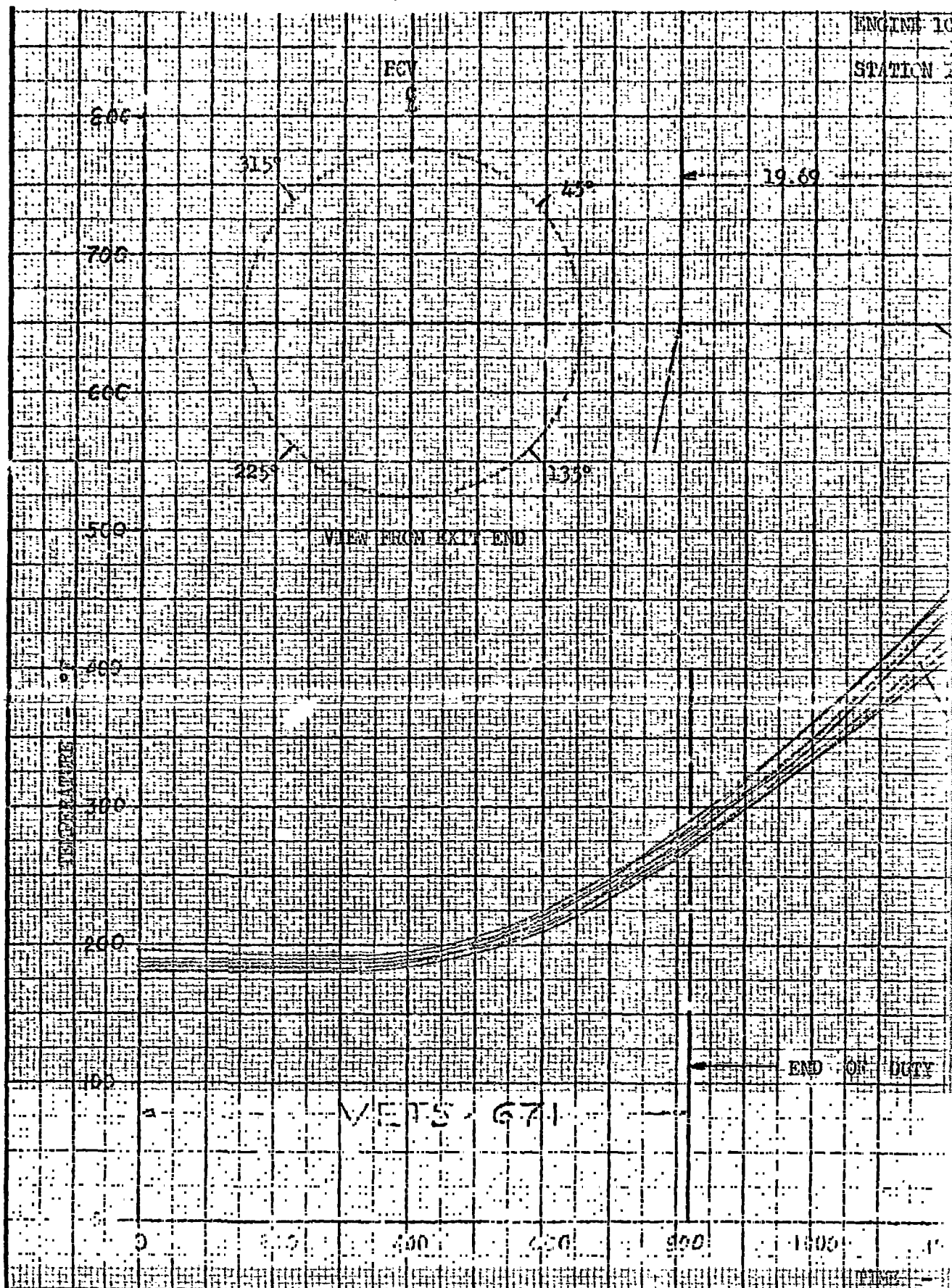
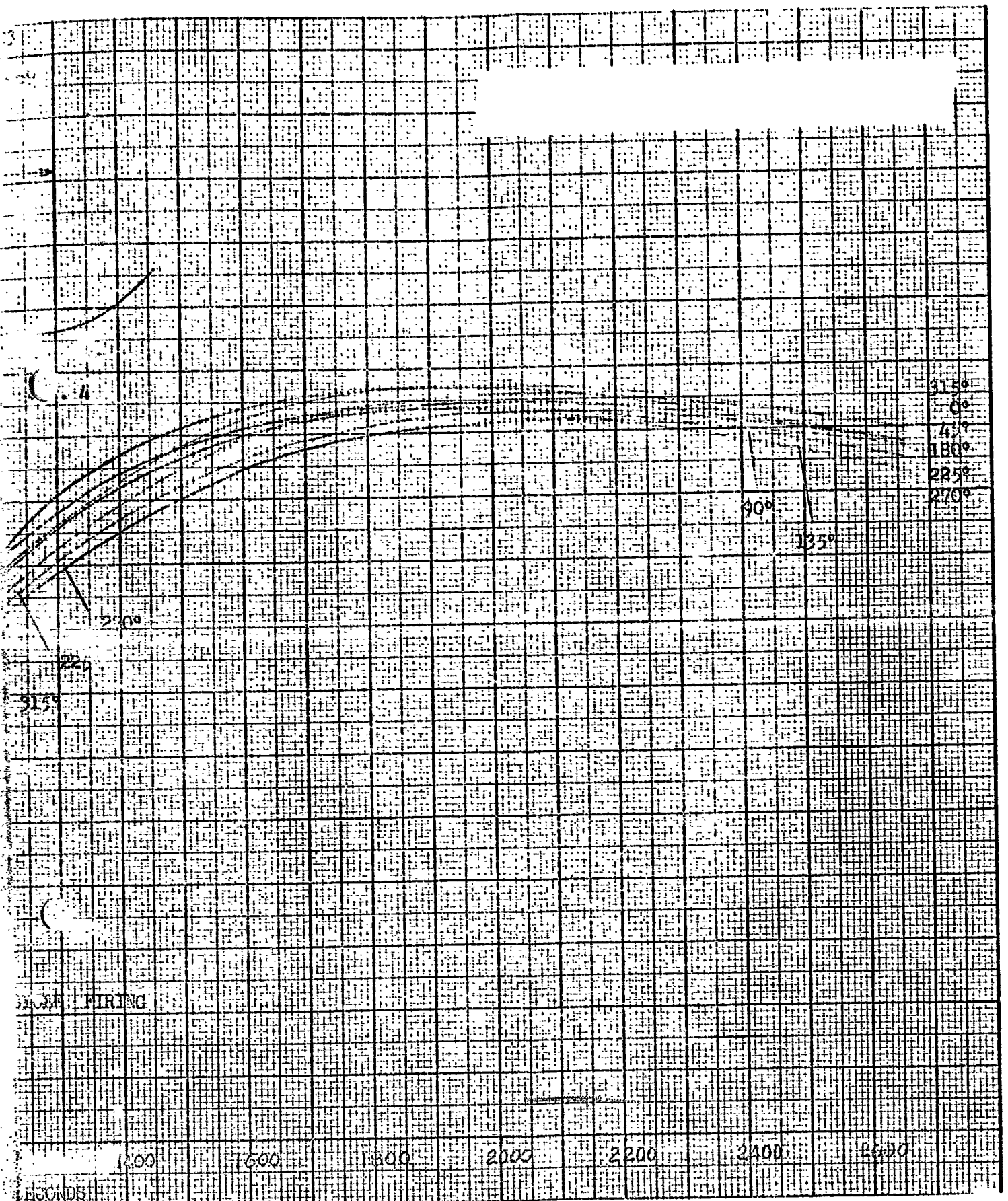


Figure 5-9. Titanium Case Temperature Data



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Figure 5-10. Titanium Case Temperature Data

<u>Test Description</u>	<u>Thrust (%)</u>	<u>Duration (sec)</u>
Duty Cycle	50	66
(Phase II)	40	30
Continued	25	46
	15	25
	40	44
	25	162

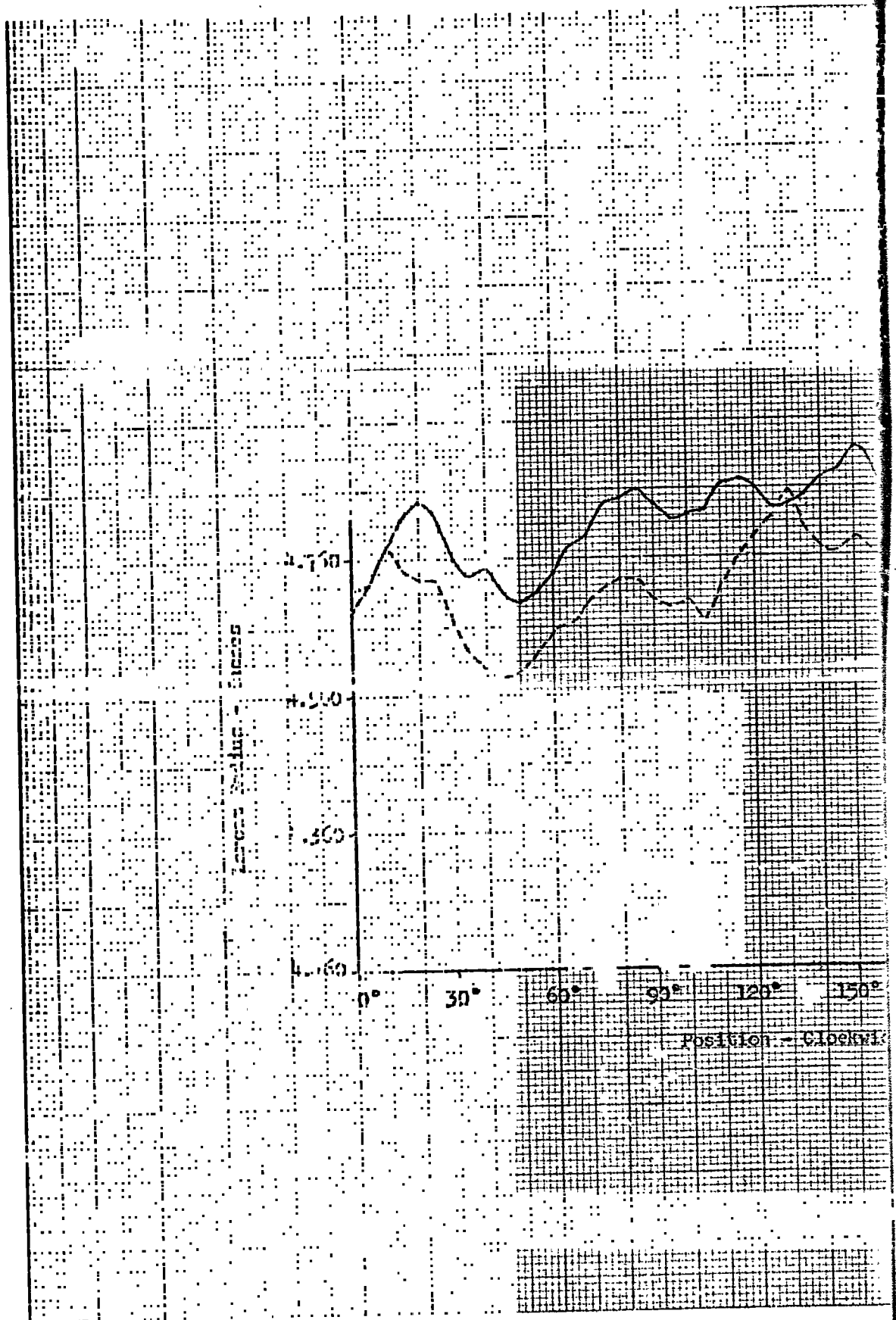
The erosion during the 320 second FTP burn was 2.7 percent based on the difference between peak c^* (characteristic velocity) and the c^* at the end of the FTP burn. The total throat area at the end of the FTP burn was only 0.5 percent more than the original throat area or, in other words, the throat shrank 2.2 percent before it started increasing in area.

Beyond the full thrust point, throat erosion rates computed from c^* are quite erratic, but the total measured throat increase of 28.2 percent indicates that the erosion rate was very great when the engine was throttled down. The reason for this phenomenon is unclear since, as was predicted from heat transfer tests of the A contour, a significant reduction in erosion was achieved at the full thrust level. However, the erosion over the throttling thrust range was higher than with the standard LMDE Qual B contour chamber, contributing to an overall higher erosion as shown in the erosion plots of Figure 5-11.

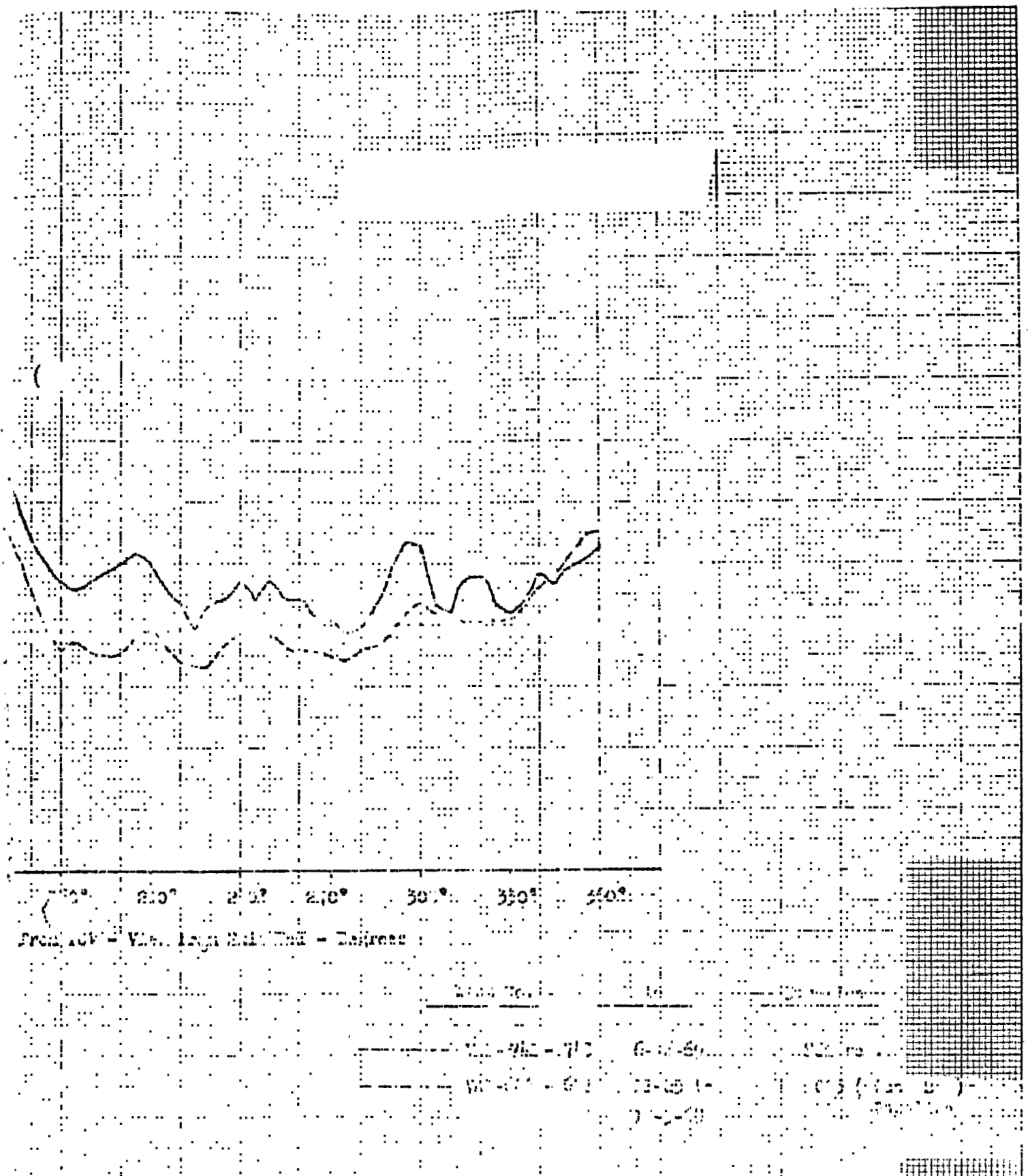
The post-firing condition of the ablative chamber is shown in Figures 5-12 through 5-14.

Although the MX-2600 erosion with Contour A was much greater than in either the baseline (Section 4.2) or Contour C composite firings, (Section 5.3) it will be noted that the remaining char, Figures 5-15 and 5-16, when compared with Figure 5-6, was slightly less at the same stations in the entrance and throat regions. The thickest char was located about four inches upstream compared to Contour C.

As a result of this test, further evaluation of Contour A was discontinued. Nevertheless, the fact that its lower heat flux at FTP was confirmed by reduced erosion over that of Qual B in both streak chambers and ablative silica throats leads to the conclusion that, should the need arise for a mission duty cycle conducted primarily at full-thrust, a Contour A chamber would certainly offer a distinct advantage in greatly reduced erosion and the concomitant overall performance.



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Figure 5-11. Contour A Vs. Qual B Erosion

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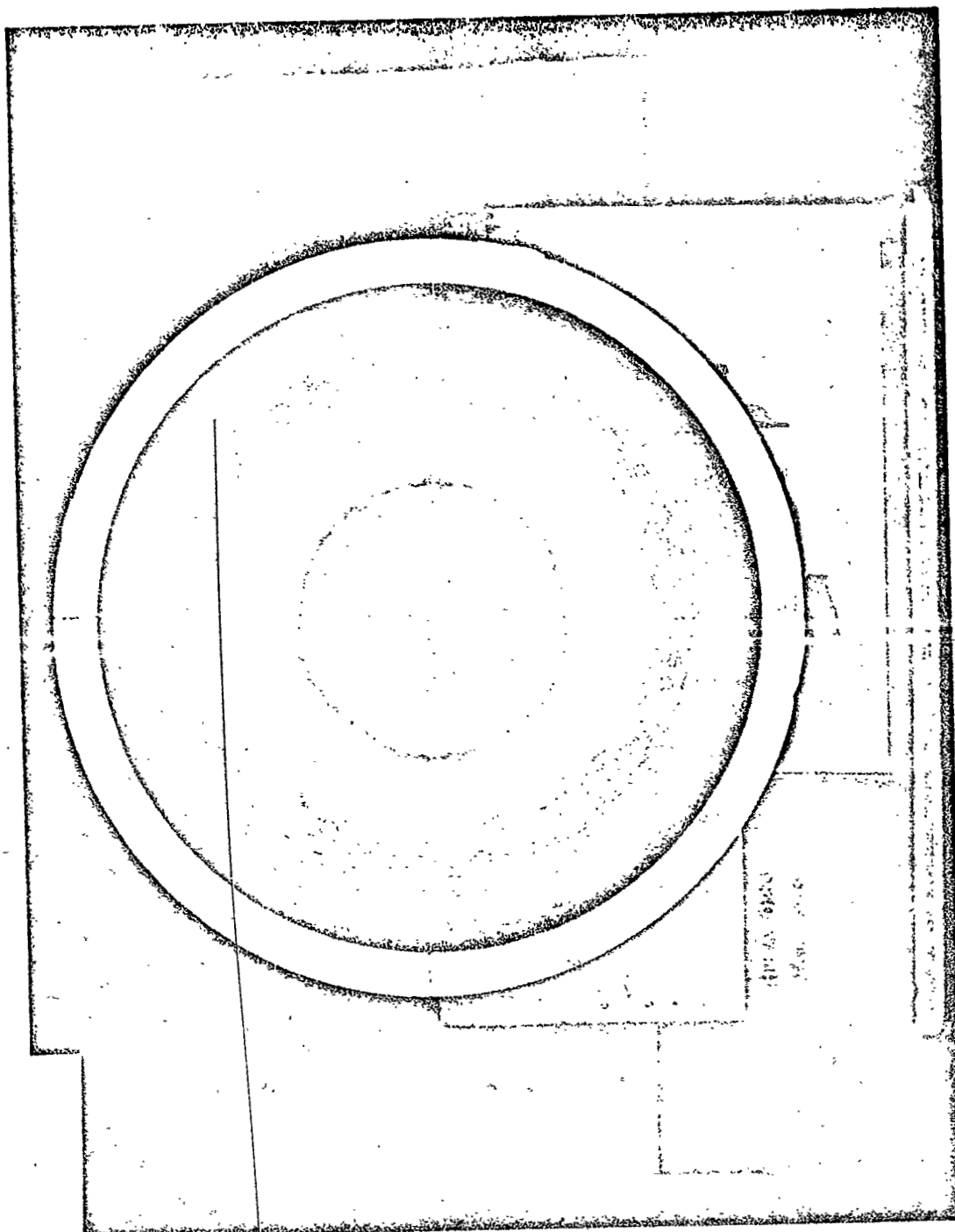


Figure 5-12. Post-Test Condition of 2:1 Ablative Contour A Chamber (View Looking Aft)

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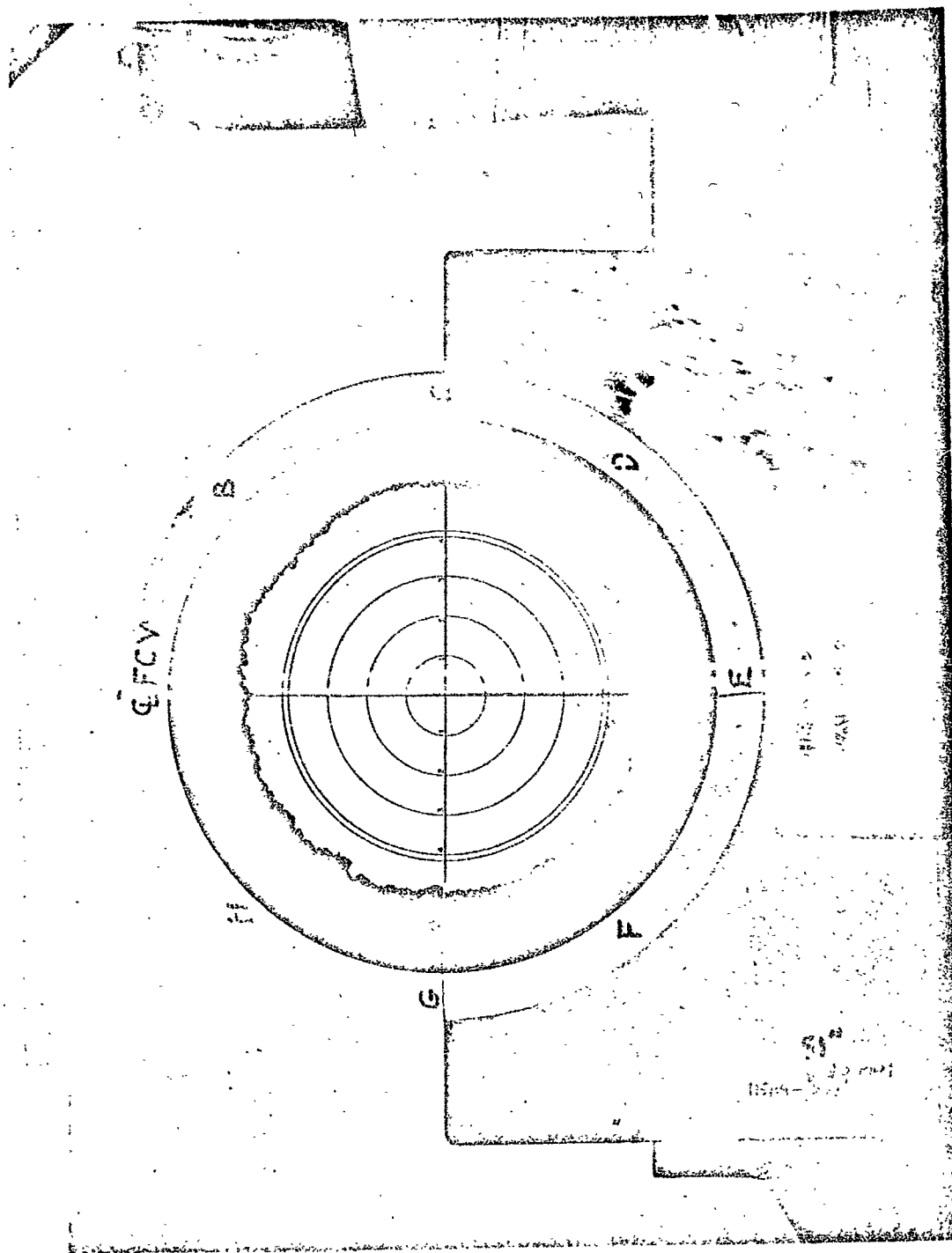


Figure 5-13. Post-Test Condition of 2:1 Ablative Contour A Chamber (View Looking Forward)



Figure 5-14. Post-Test Condition of 2:1 Ablative Contour A Chamber (View Looking Aft)

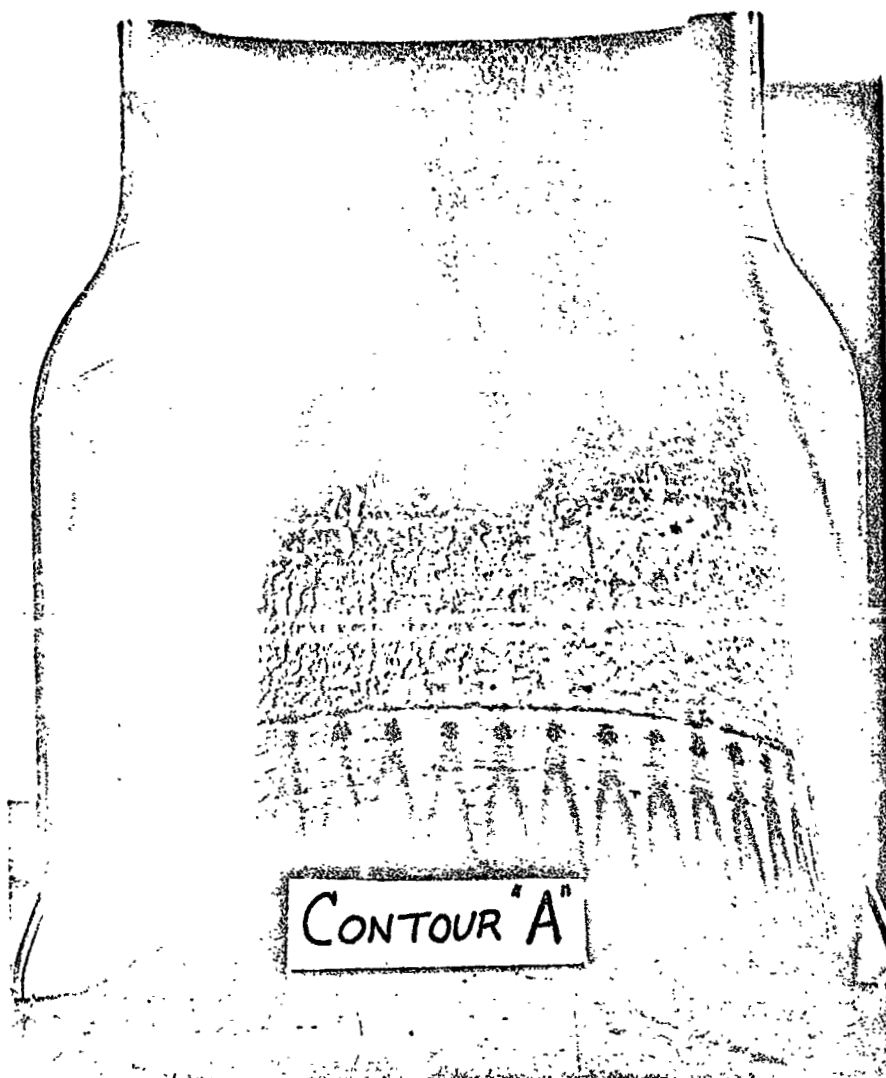


Figure 5-15. Liner Char Pattern After
Tests VA1-741, 742 and 743

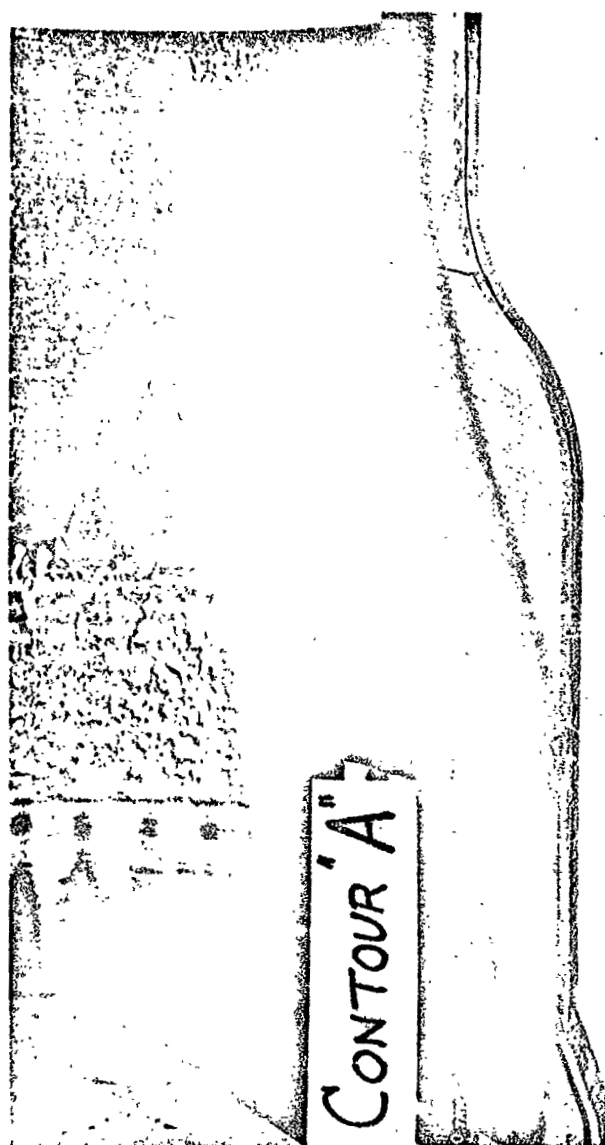


Figure 5-16. Contour A Liner Char Pattern
After Tests VA1-741, 742 and 743

5.3 ADVANCED MATERIALS COMPOSITE THROAT

Based on results and recommendations emanating from the Materials Survey and Evaluation Phase of the program, three ablative materials were selected for reactive testing. These were combined into the quadrants of a composite throat for direct comparison with the LMDE standard MX-2600 silica-fabric/phenolic. In addition to the advantage gained by direct comparison of materials in the same reactive test, a considerable cost saving resulted by molding three of the four simultaneously in a billet mold. The four quadrants, shown in Figure 5-17, consisted of the following phenolic laminates:

- MX-2600, with the current LMDE throat silica-fabric and a silica/carbon ratio of 4.5
- MXQ 190, with a quartz fabric and a silica/carbon ratio of 4.5
- MXQ 190 of high-resin content, precharred and reimpregnated, with a silica/carbon ratio of 1.1 (TRW-12A)
- A composite quartz and graphite fabric, with a silica/carbon ratio of 1.0 (TRW-2A)

As discussed in Section 2., quartz fabric offered a potential improvement over silica. The third and fourth quadrants represented lower silica-to-carbon char ratios aimed at promotion of silicon carbide formation in situ during the test firing. The third quadrant was similar to the TRW-12 laboratory specimen, and the fourth quadrant represented the TRW-2A torch test specimens. Silicon carbide was formed, and little or no dimensional changes occurred when those specimens were heated in laboratory tests.

The chamber configuration chosen for the test series was Contour C that exhibited the best c^* performance in the heat-sink and water-cooled chamber tests and whose throat heat flux was slightly less than that of the Qual B configuration over the throttle range. Figure 5-18 shows the Contour C slotted MX-2600 turbulence ring. The slotted ablative face plate is shown in Figure 5-19. The slots provide for channeling barrier coolant farther downstream than is possible with the Qual B chamber.

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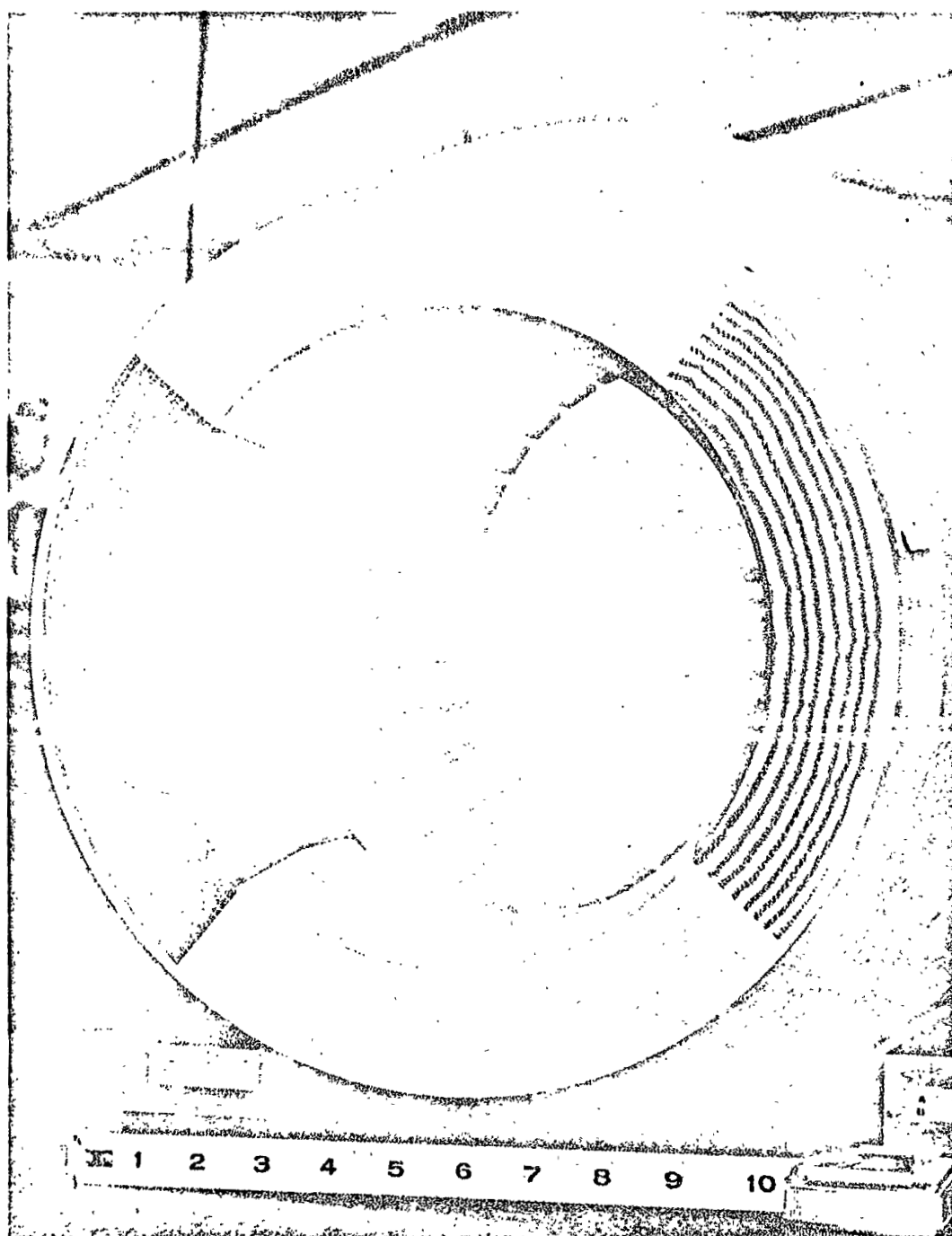


Figure 5-17. Throat Erosion Evaluation Liner
Viewed from Aft End - Clockwise
from Bottom: MX-2600, TRW-12A,
MXQ-190, TRW-2A

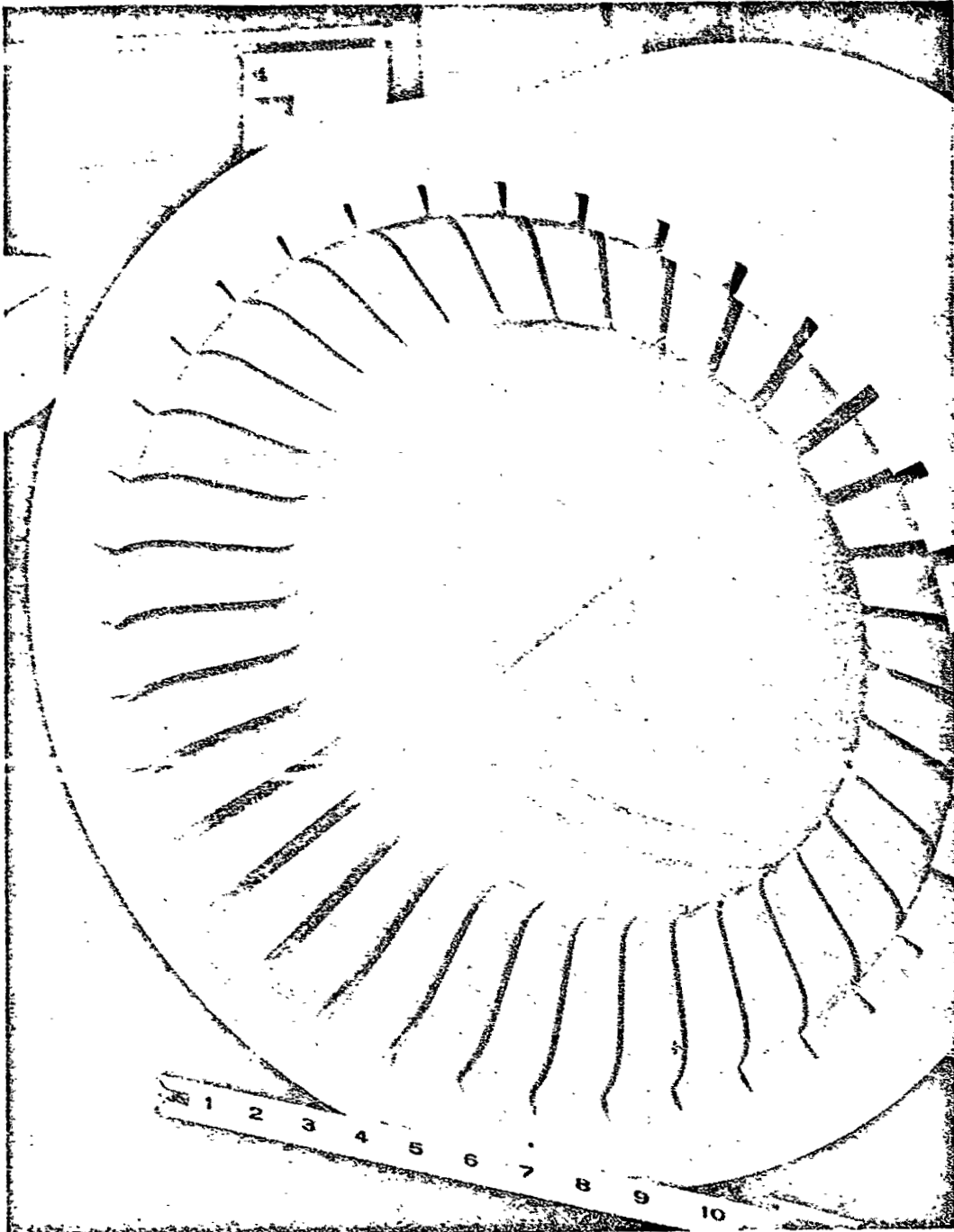
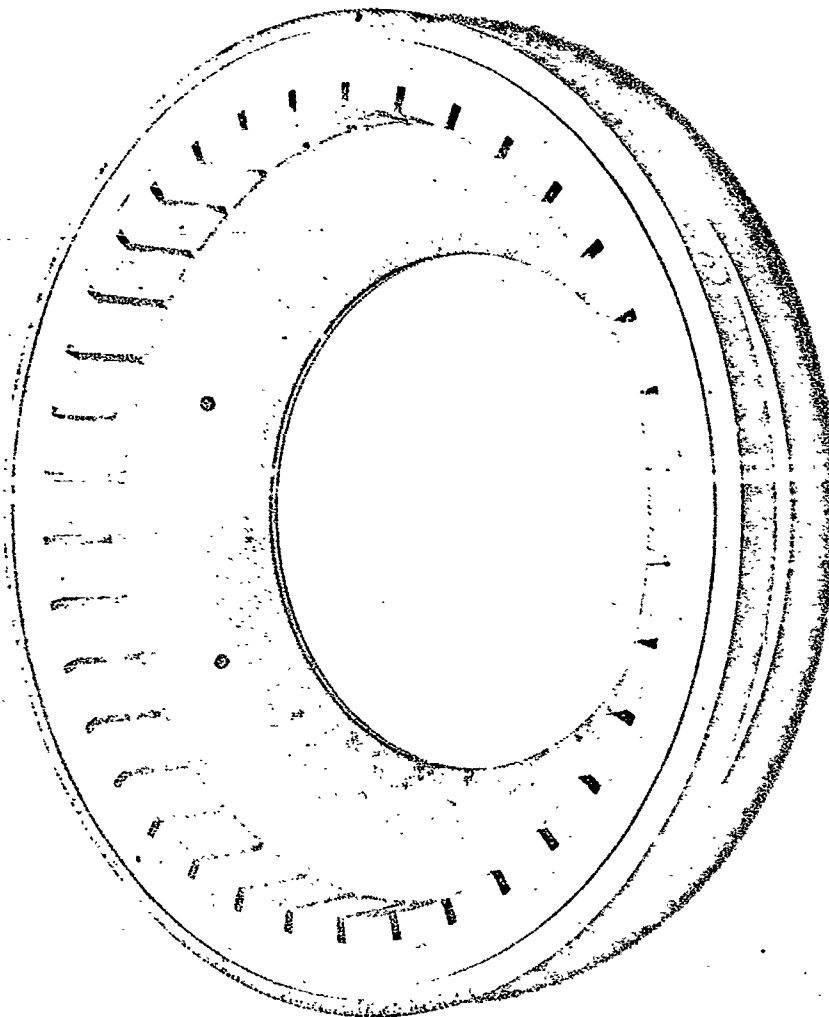


Figure 5-18. Throat Erosion Evaluation Chamber Liner
with Slotted Turbulence Ring - Viewed
From Forward End



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Figure 5-19. Ablative Faceplate Modified to Include Barrier-Coolant Channels

The duty cycle is summarized in Table 5-4. Test VA1-733 simulated a normal LMDE acceptance test; test VA1-734 simulated the orbit injection portion of a duty cycle; and test VA1-735 simulated the braking phase, 320 seconds of which were at full thrust. At this point, the engine was removed from the stand for a throat measurement. Its condition is shown in Figure 5-20.

The test series was completed with VA1-736, a checkout test, and VA1-737, simulating the flare-out, hover, and landing phases of a duty cycle. Test VA1-737 included 60 seconds at FTP before throttling down to induce the thermal gradient into the liner that would exist normally at that point had the firing not been interrupted for the throat measurement. The post-firing condition of the chamber is shown in Figures 5-21 and 5-22.

The throat erosion profiles measured after tests VA1-735 and VA1-737 are compared with the Qual B baseline erosion (Section 5.1) in Figure 5-23. The shapes are generally similar except for discontinuities along the bondlines of the four-quadrant throat. To eliminate bondline interaction effects, computed erosion of the middle 60-degree segments of each quadrant is presented in Table 5-5. It will be noted that at full thrust the precharred and reimpregnated quartz-fabric/phenolic (TRW-12A) and the standard quartz-fabric/phenolic (MXQ-190) had about half of the erosion of the standard silica-fabric/phenolic (MX-2600), even though the former were located in more erosion-inducing regions. This trend prevailed through the low-thrust portion of the test series. In the final analysis, the effect of injector variation was compensated for by comparing erosion in each quadrant with that experienced in that quadrant during the Qual B Baseline Erosion Duty Cycle Test (Section 5.1). This new standard of comparison results in the final line of Table 5-5 where certain facts become obvious.

- The materials evaluation test was a valid comparison in that MX-2600 eroded almost exactly the same amount as it did in the baseline test.
- TRW-2A eroded 20 percent more than MX-2600.

TABLE 5-4
ADVANCED MATERIALS EVALUATION DUTY CYCLE

<u>Test No.</u>	<u>Thrust (Percent)</u>	<u>Firing Duration (Sec)</u>	<u>Cumulative Firing Time (Sec)</u>
VA1-733	15	3	
	FTP	7	
	50	6	
	25	7	
	15	<u>3</u> 26	26
VA1-734	15	26	
	FTP	7	
	25	<u>6</u> 39	65
VA1-735	15	26	
	FTP	320	
	15	<u>6</u> 352	417
VA1-736	15	3	
	FTP	7	
	50	6	
	25	7	
	15	<u>3</u> 26	443
VA1-737	15	30	
	FTP	60	
	60	50	
	65	48	
	50	66	
	10	30	
	25	46	
	15	25	
	40	44	
	25	<u>162</u> 561	1,004

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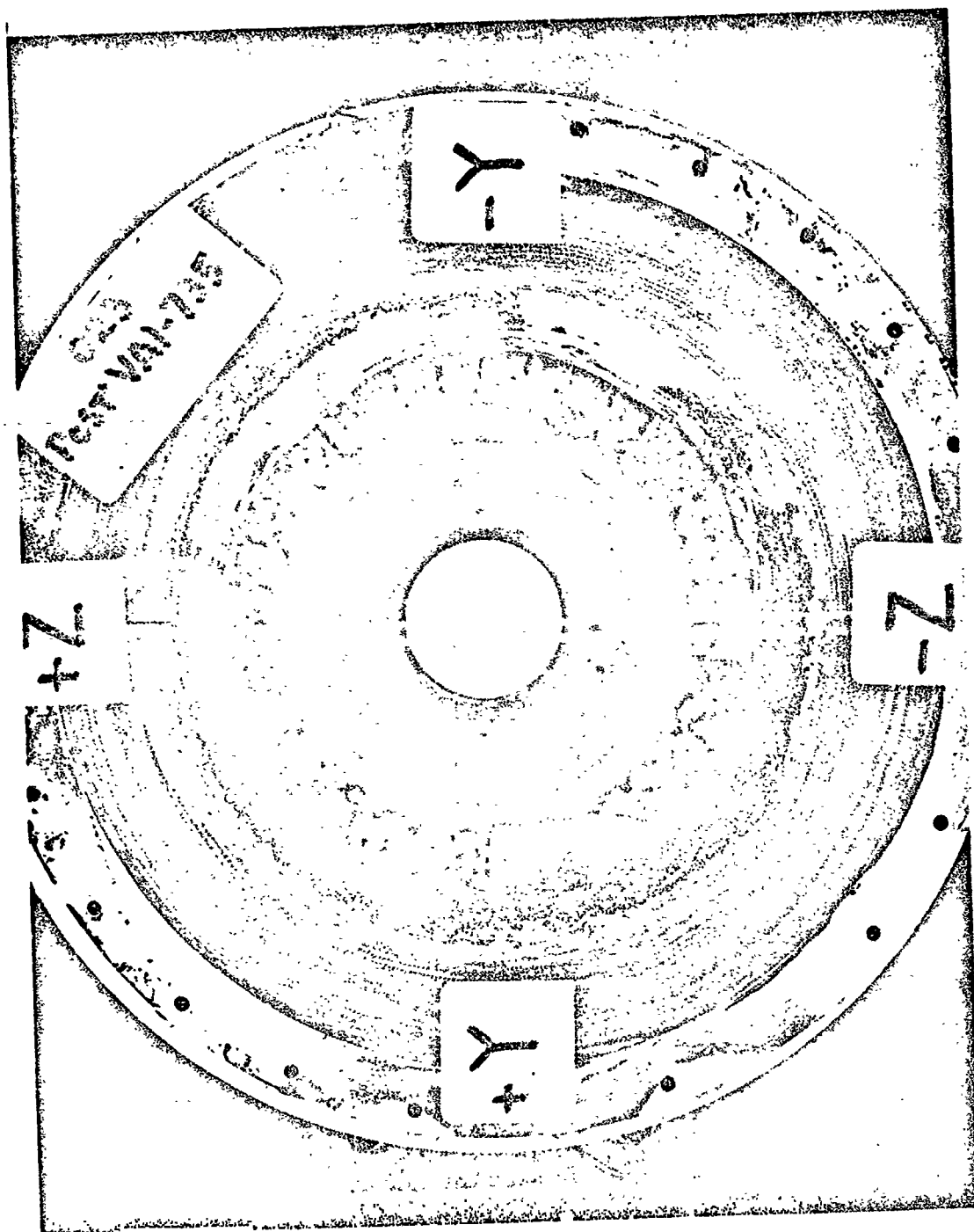


Figure 5-20. Throat Condition after Full-Thrust
Portion of Duty Cycle

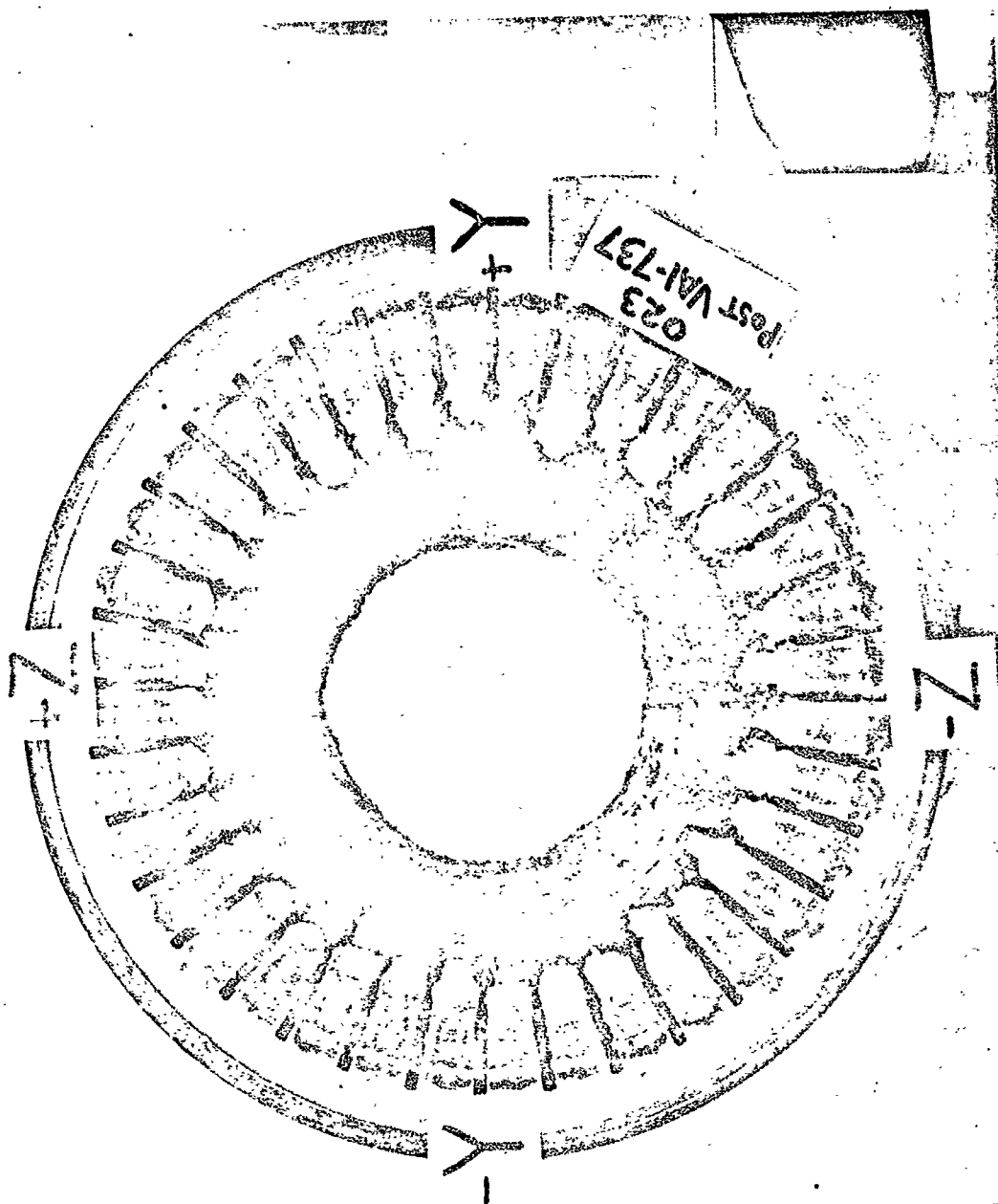


Figure 5-21. Throat and Turbulence Ring
Condition After Duty-Cycle Test

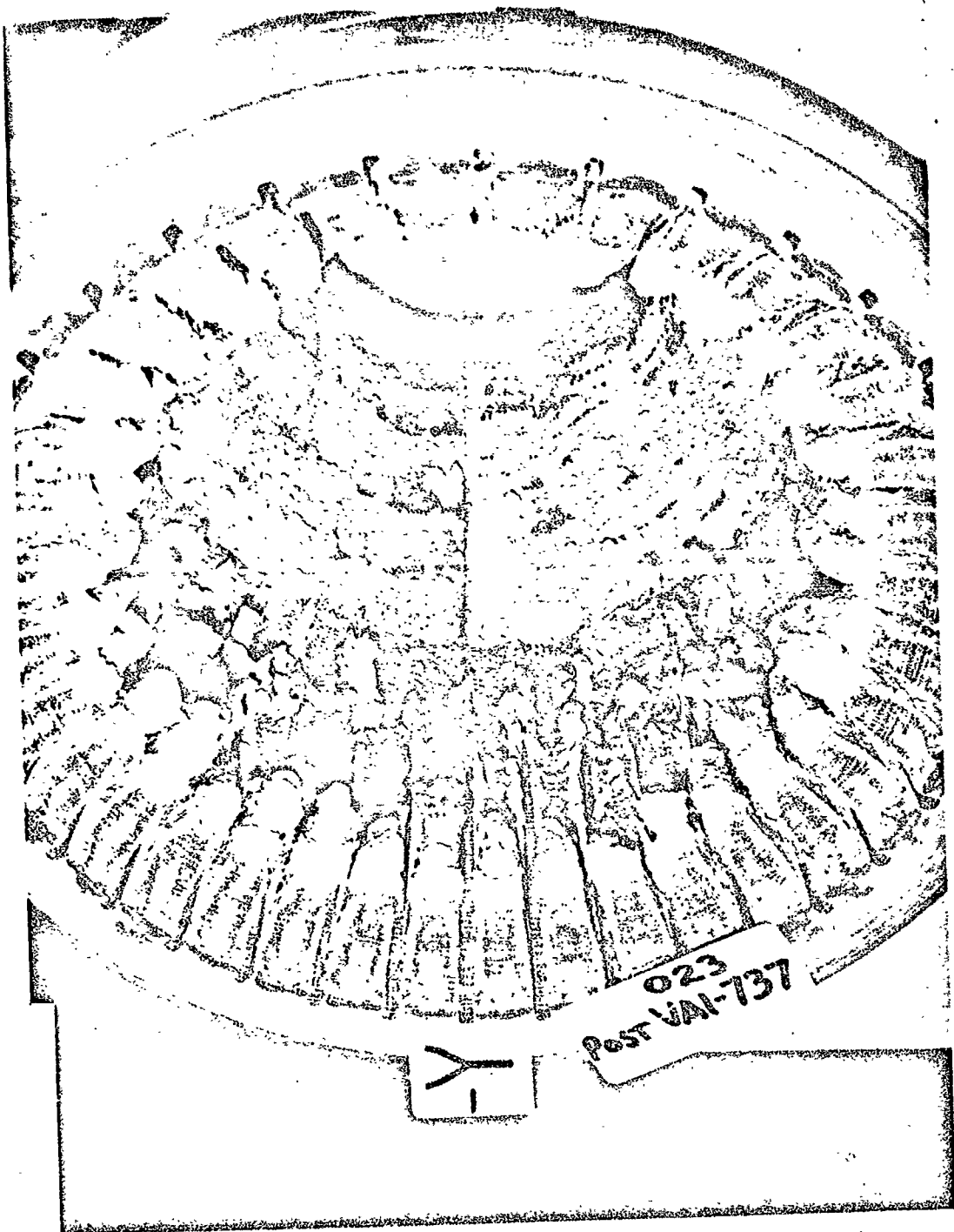
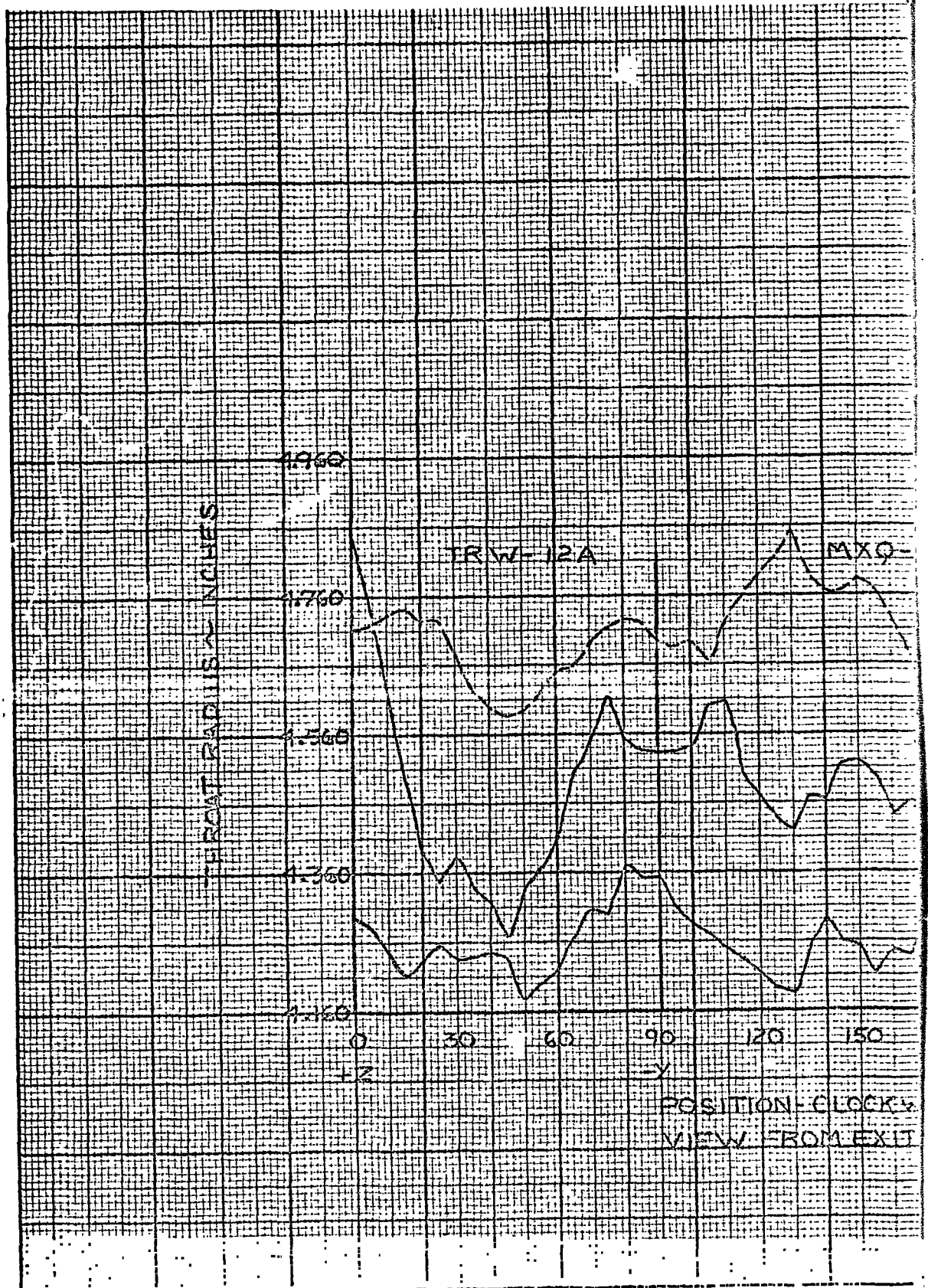
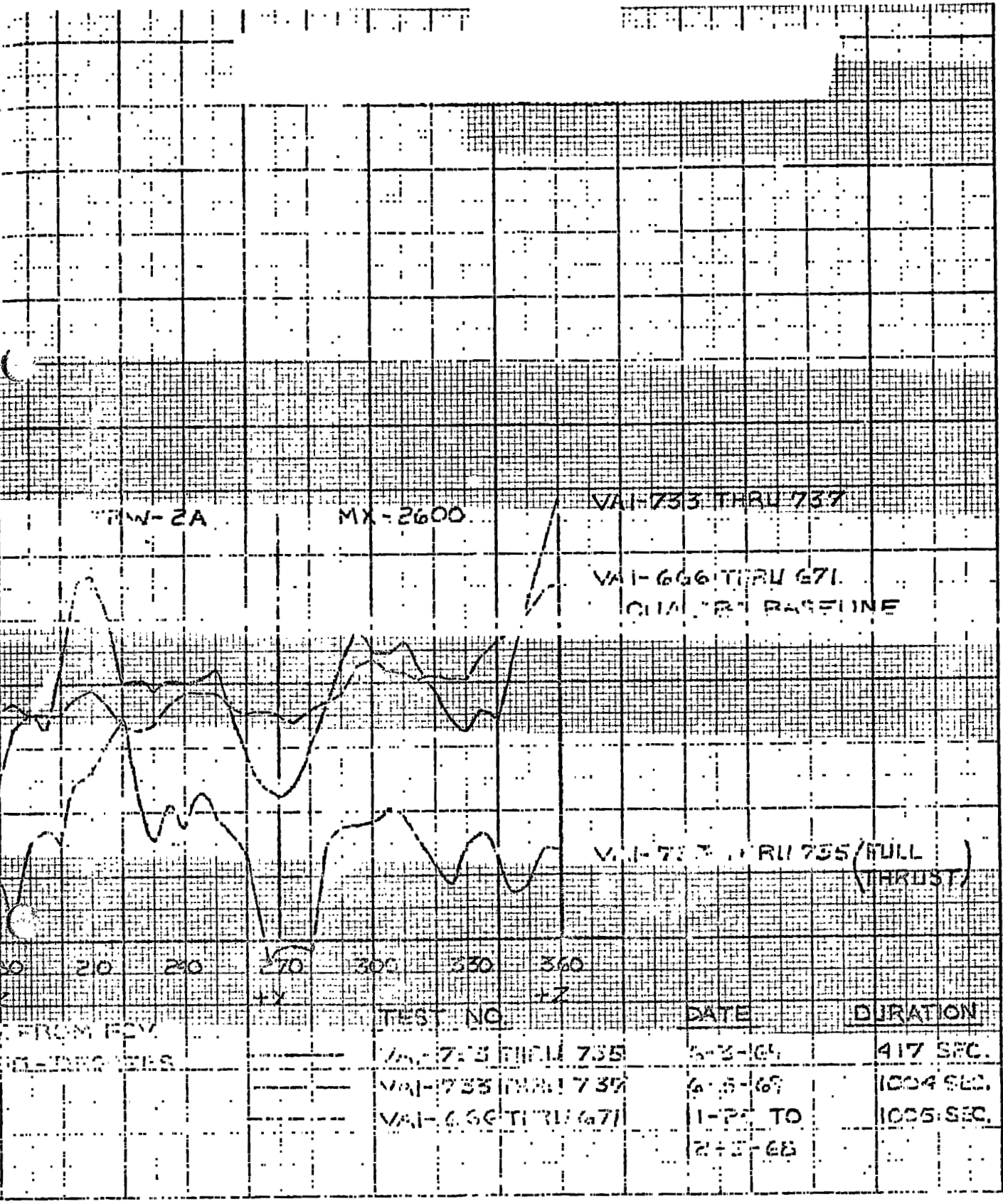


Figure 5-22. Chamber Liner Condition After Duty-Cycle Test



FOLDOUT FRAME



FOLDOUT FRAME 2 Figure 5-23. Advanced Materials Erosion Profile 5-32

TABLE 5-5
ADVANCED MATERIALS EROSION TEST SUMMARY
PERCENT THROAT AREA INCREASE

TEST NO.	TEST DURATION (SEC)	MAIN THRUST LEVEL	POSITION CLOCKWISE FROM FCV - VIEW FROM EXIT END (DEGREES)			
			15-75 (TRW-12A)	105-165 (MXQ-190)	195-255 (TRW-2A)	285-345 (MX-2600)
VA1-733 THRU 735	417	FULL	4.0	3.9	10.9	7.4
VA1-733 THRU 737	1004	DUTY CYCLE	12.5	16.8	21.8	20.4
VA1-665 THRU 671 (QUAL B BASELINE)	1014	DUTY CYCLE	25.5	30.9	18.2	21.2
EROSION AS PERCENTAGE OF BASELINE EROSION			49.0	54.4	119.8	96.2

- TRW-12A and MXQ-190 had half the erosion of MX-2600.

TRW-12A, described above, and TRW-2A, a composite phenolic laminate containing one ply of graphite fabric to each two plies of quartz fabric, were formulated with the objective of forming SiC in situ during the full-thrust portion of the firing. The SiC presence should then have decreased the low-thrust erosion.

X-ray diffraction analyses were performed on specimens taken from the surface and 1/16 inch beneath the surface from two locations; location 1 at the throat, and location 2, about 4-1/2 inches upstream from location 1. Results are shown in Table 5-6.

Although quantitative assessment of the SiC content is not possible from these results, it is estimated that the amount of SiC may be as high as 10% by volume in the surface layer of all specimens. No SiC was detectable 1/16 inch beneath the surface except in the TRW-2A specimen in the upstream location 2 where weak intensity SiC prevailed. It is interesting to note that SiC did not form in TRW-12A specimen during the pyrolysis cycle prior to engine firing and that SiC formation was entirely the result of reaction of silica and carbon in the chamber environments.

It is not clear at this time whether the formation of SiC is beneficial, but certain observations are apparent:

- The greatly improved erosion resistance of MXQ-190 (over that of MX-2600) and its equivalence to TRW-12A does not warrant the additional expense of precharring and reimpregnating to form the latter composition.
- The increased erosion and thermal conductivity induced by the addition of graphite fabric to the quartz-fabric/phenolic laminate precludes use of TRW-2A in the LMDE environment.

The char pattern at the center of each quadrant is shown in Figure 5-24. It will be noted that the structural integrity of the TRW-12A was good, even though it was severely delaminated before the firing test.

Table 5-6. X-RAY DIFFRACTION RESULTS

Specimen	Location 1		Location 2	
	(Surface Layer 1/16")	(Second Layer 1/16")	(Surface Layer 1/16")	(Second Layer 1/16")
TRW-2A	SiO ₂ (α-Quartz) α-SiC	SiO ₂ (α-Quartz)	SiO ₂ (α-Quartz) α-SiC	SiO ₂ (α-Quartz) α-SiC (weak)
TRW-12A Prior to Engine Firing		(No crystalline materials de- tected)		
TRW-12A	SiO ₂ (α-Quartz) α-SiC	SiO ₂ (α-Quartz)	SiO ₂ (α-Quartz) α-SiC	SiO ₂ (α-Quartz)
MXQ 190	SiO ₂ (α-Quartz) α-SiC	SiO ₂ (α-Quartz)	α-SiC	SiO ₂ (α-Quartz)
MX-2600	SiO ₂ (α-Quartz) α-SiC	SiO ₂ (α-Quartz)	α-SiC	SiO ₂ (α-Quartz)

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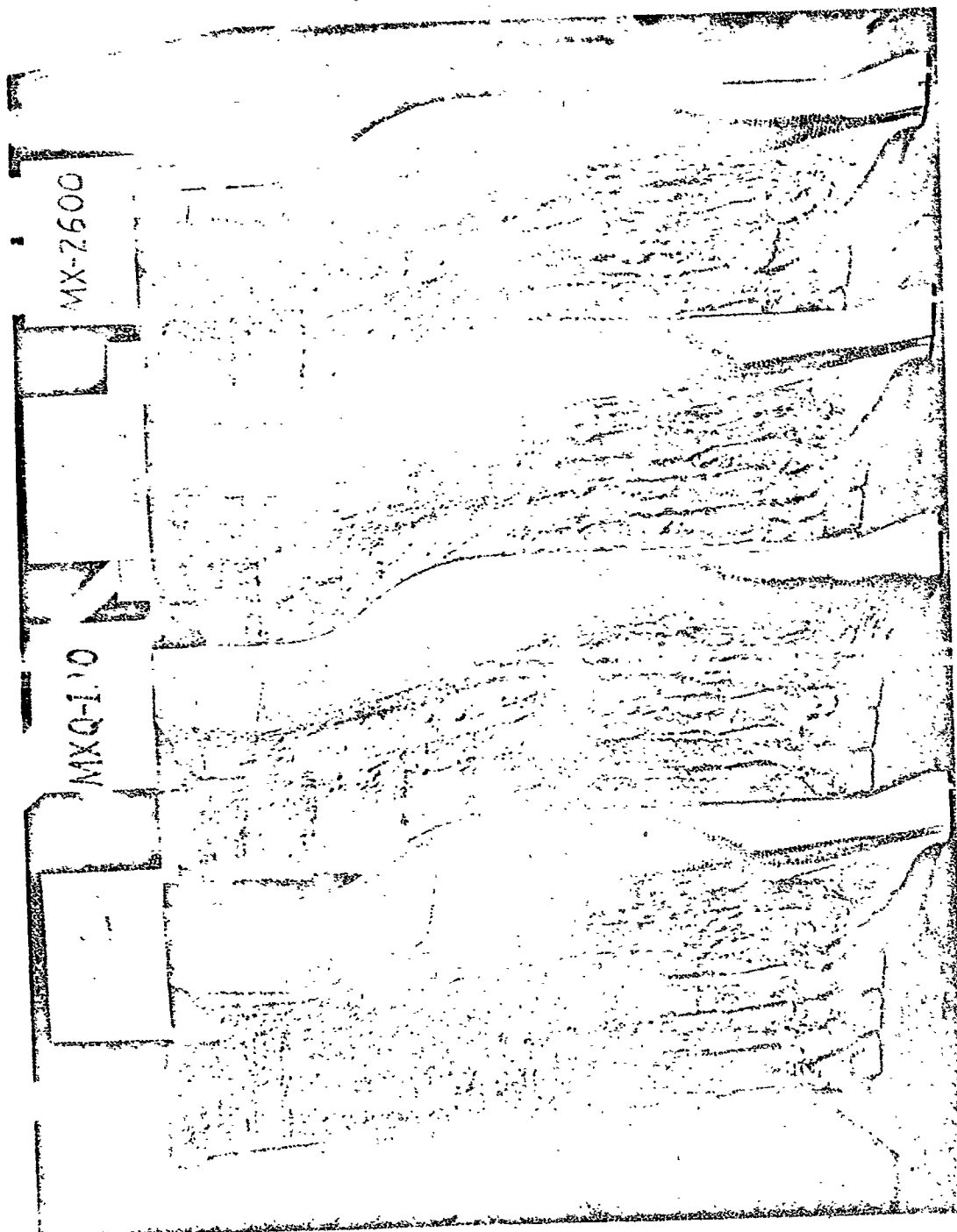


Figure 5-24. Advanced Materials Erosion Evaluation
Chamber Liner Char Patterns

The MXQ-190 had one delamination at the throat compared to several in the MX-2600, and had about 1/8 inch more virgin material remaining. The TRW-2A was charred through completely and had multiple delaminations. There was evidence of gas flow through one delamination to the glass-fabric/phenolic overwrap.

The char pattern is compared to that of the Qual B Baseline (Section 5.1) in Figure 5-25. It will be noted that the highest eroding region of the chamber has been shifted 2-1/2 inches upstream in the Contour C configuration. This deeply eroded region is shown more clearly in Figure 5-26. Its occurrence is even more surprising considering the fact that the barrier coolant slots have retained their original configuration in the same region. The extreme erosion of the MX-2600 turbulence ring is depicted in the comparison with the original contour in Figure 5-27. A quartz-fabric/phenolic construction should alleviate this condition.

The faceplate was in excellent condition after the test series, as may be noted in Figure 5-28.

The increase in C^* for the C contour chamber over that of the Qual B configuration observed in the heat sink chamber tests was further verified in the composite chamber duty cycle testing.

Based on the titanium shell backwall temperature response during VA1-637, the throttled low-thrust portion of the duty cycle test, the following thermal diffusivity ratios were computed for each throat quadrant material relative to MX-2600 silica-fabric/phenolic:

<u>Material</u>	<u>α/α MX-2600</u>
MX-2600	1.00
MXQ-190	0.98
TRW-12A	1.61
TRW-2A	2.54

The ratios presented are the mean values computed by two different methods as discussed in Reference 35.

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Figure 5-25. Comparison of Contour C Char Pattern with Qual B Baseline.

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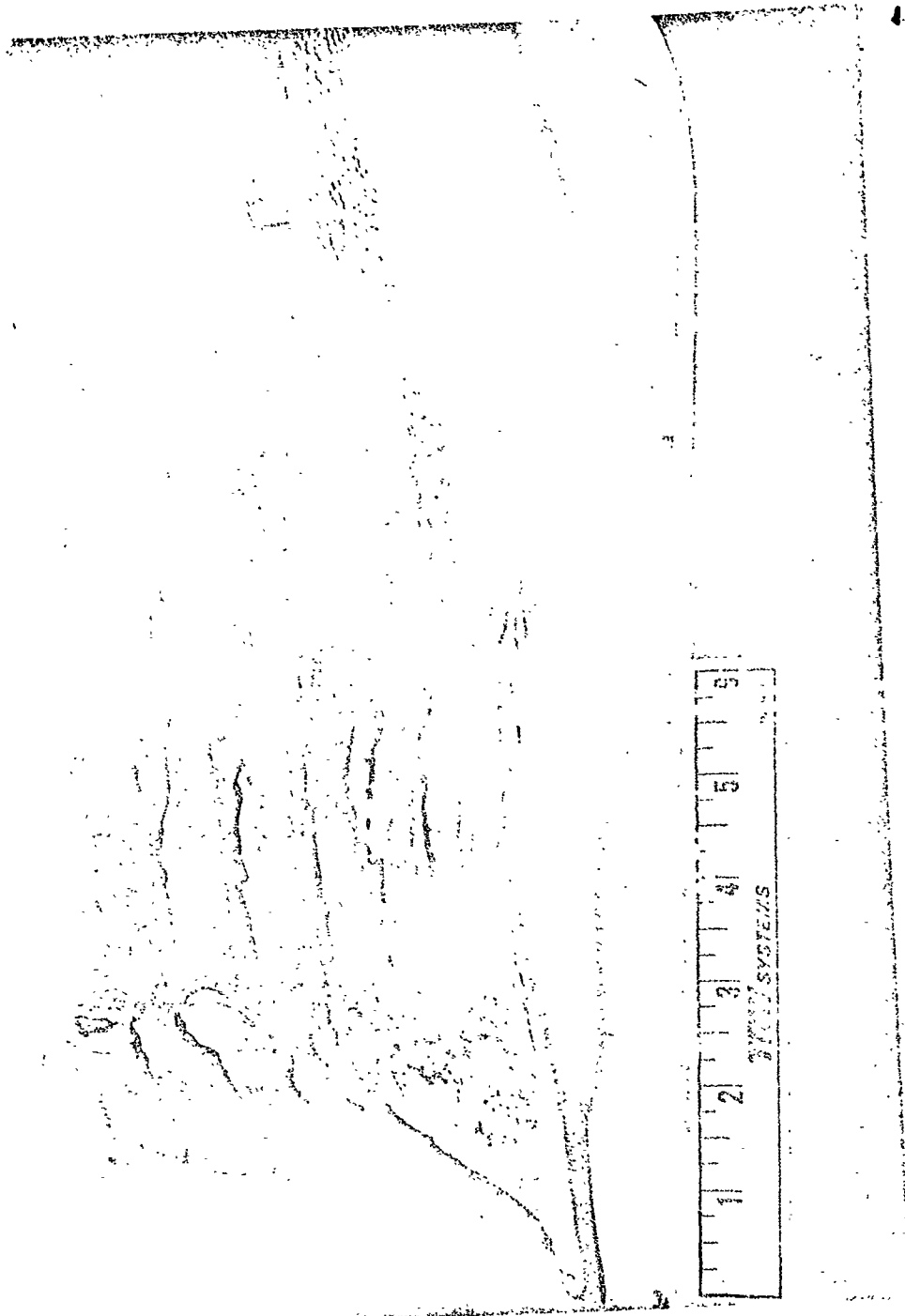


Figure 5-26. Turbulence Ring and Chamber
Liner Erosion

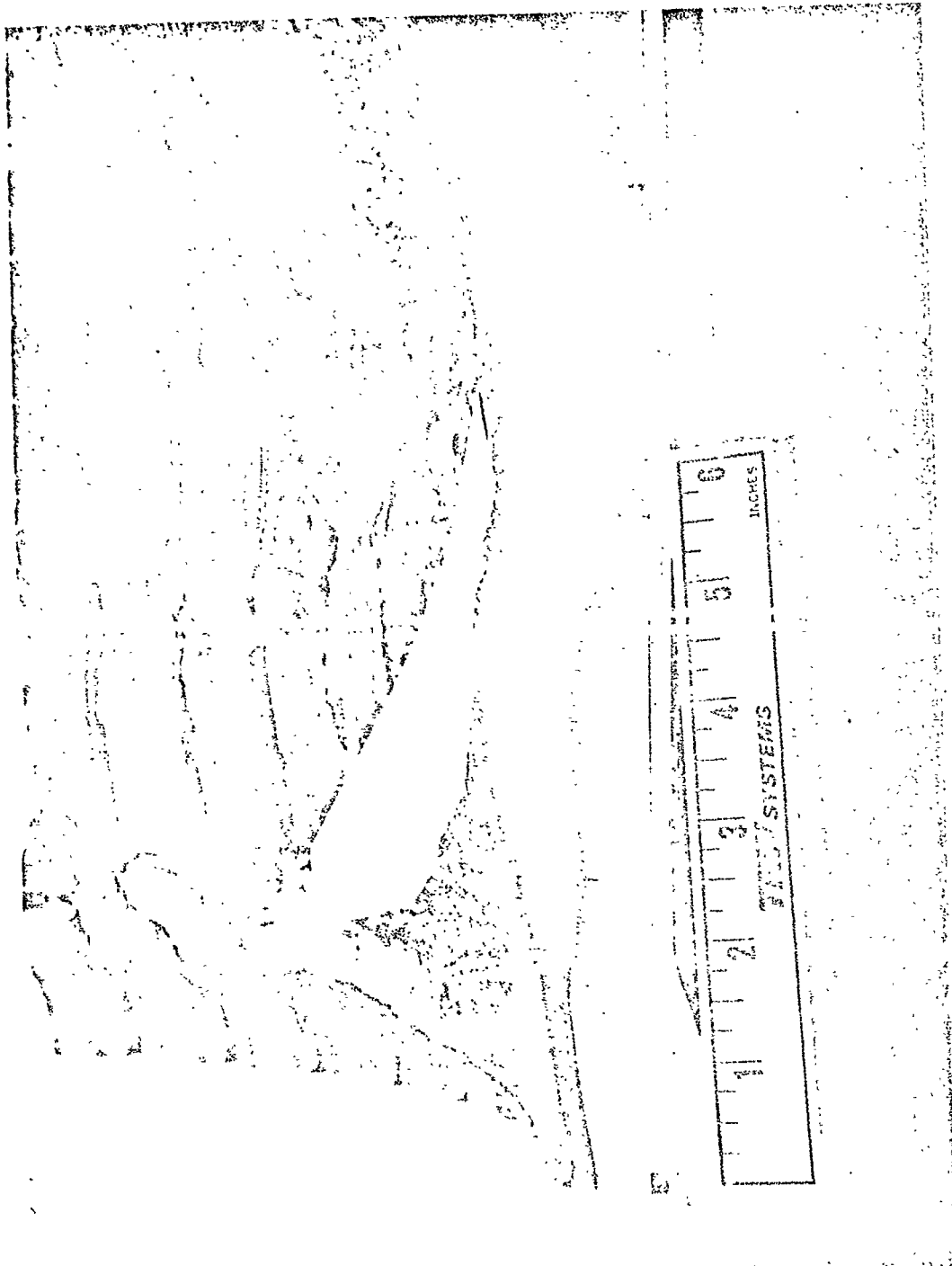


Figure 5-27. Comparison of Turbulence Ring Erosion with Original Contour

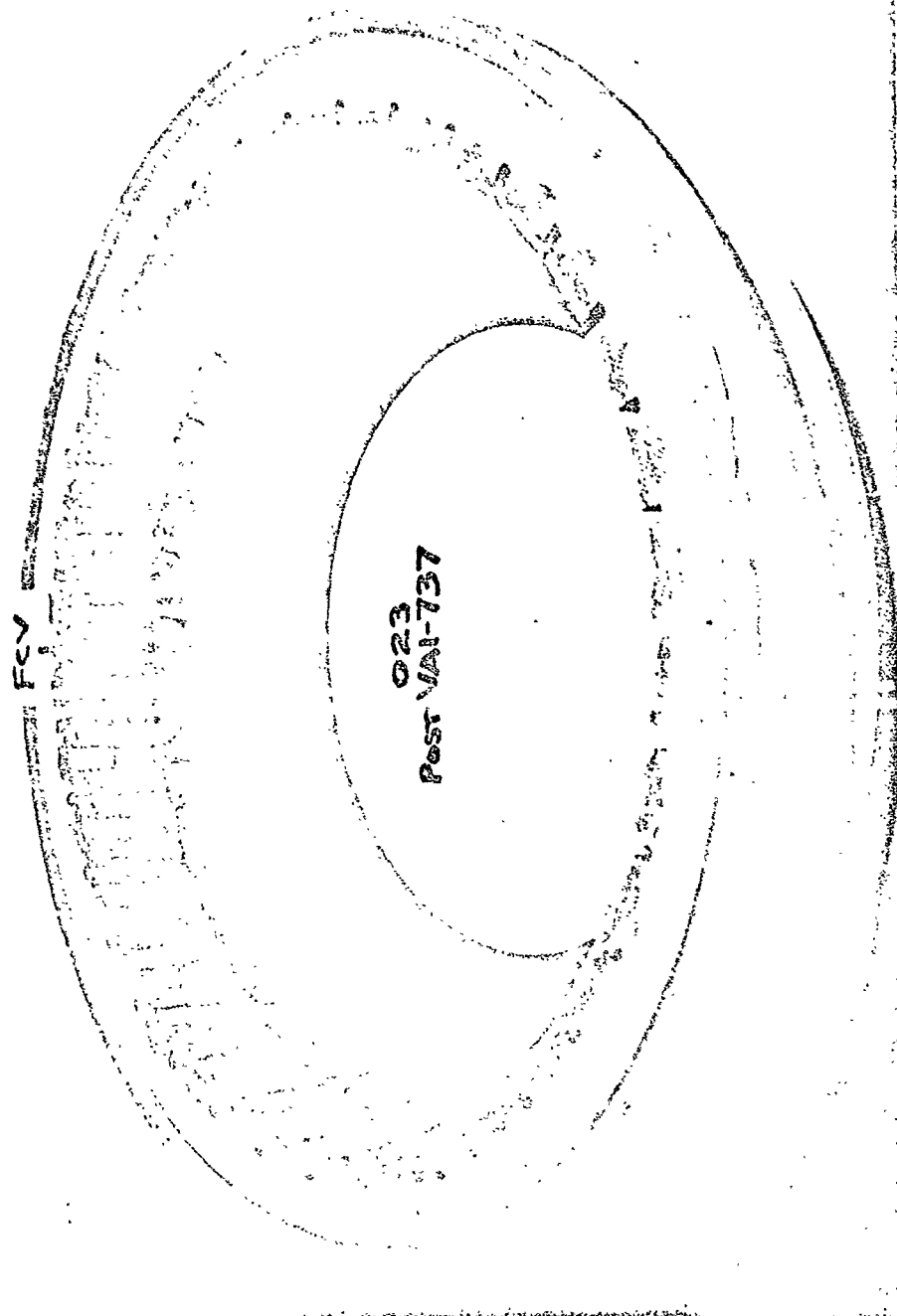


Figure 5-28. Slotted Ablative Faceplate After Duty-Cycle Firing

Based upon the results of the advanced materials erosion test, it was concluded that the quartz-fabric/phenolic resin laminated material would achieve a significant reduction in erosion of up to 50 percent over the current MX-2600 silica-fabric/phenolic without further materials development effort.

The precharred quartz fabric/high-resin-content phenolic showed an even greater reduction in erosion. Although it is believed that this material has a higher potential for erosion reduction, the development time and cost for optimization of the formulation and processing was beyond the scope of the CIC program. Therefore, the quartz-fabric/phenolic material was selected for evaluation in an advanced prototype LMDE combustion chamber.

5.4 PROTOTYPE CHAMBER

5.4.1 Design and Fabrication

The original design for the Phase III prototype chamber consisted simply of a quartz-fabric/phenolic, oriented 75 degrees-to-centerline in the slotted turbulence ring and 60 degrees-to-centerline in the throat, in the Contour C configuration. The face-plate was the current LMDE-qualified silica-fabric/elastomer-modified phenolic in a Convolay configuration.

However, at meetings with Grumman and NASA MSC personnel in early August 1969, as a direct benefit of the CIC program, it was decided to evaluate quartz in the current LMDE Qual B configuraiton. The reason was a proposed increase in the LM tank capacity to 19,562 pounds of propellant for the extended G mission duty cycle. The increased hover capability afforded would increase the firing time by some 90 seconds above that of the nominal mission duty cycle. It was believed that the current Qual B chamber reliability would be decreased significantly thereby.

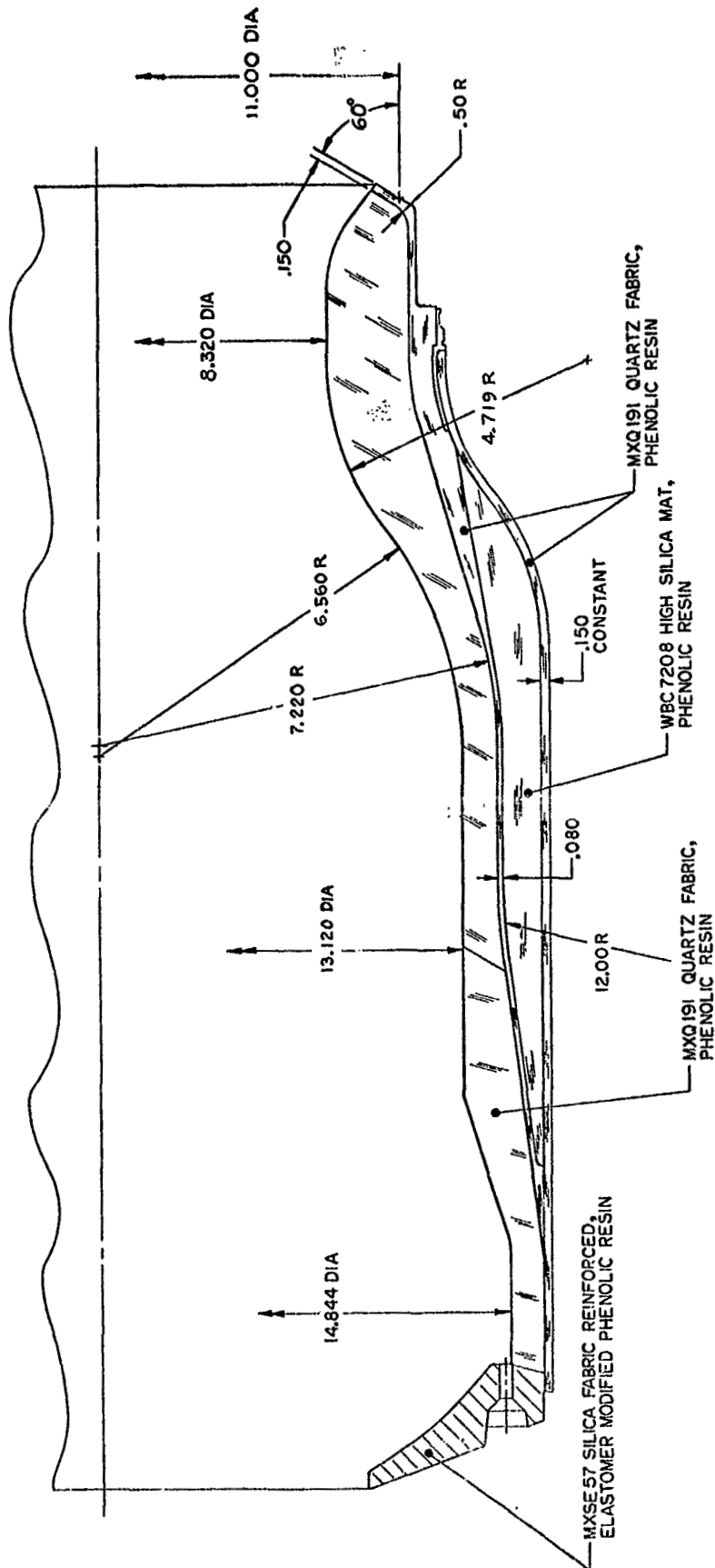
At the meeting it was decided that the Phase III prototype design should incorporate as many design features of the Qual B quartz design as possible. In this manner performance of both could be compared under

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similar reactive test conditions. In addition, a "universal chamber billet" could be fabricated that would permit final machining to either configuration, or a backup, for a DVT program that would lead to qualification testing in time for the LM-10 mission.

Within the above guidelines, and after several iterations, the final Phase III prototype design, shown in Figure 5-29, was approved by MSC. The Qual B quartz design is shown in Figure 5-30. In addition to the quartz-fabric laminate, both designs incorporated other improvements over the current (Qual A) design, shown in Figure 5-31. They include:

- (a) A quartz-fabric/phenolic tape wrap between heavyweight and lightweight backup materials to act as a gas seal should the charred inner liner delaminate.
- (b) Extension of the intermediate quartz tape overwrap into the joint region with the exit cone liner, with the actual joint moved 1/4 inch axially downstream from that of the current Qual B design. This feature offers several advantages:
 - Adds 1/4 inch more material to the chamber liner wall thickness in a critical region of the throat.
 - Overwrap feature acts as a gas seal should charred throat delaminate.
 - Overwrap extending downstream of throat is integral with cooler upstream portions of each ply so that even if the former shrinks when charred, it cannot be ejected.
- (c) A quartz-fabric/phenolic outer overwrap replaces silica-fabric/phenolic overwrap. The quartz-fabric/phenolic laminate tensile strength is 65,000 psi, compared to the silica-fabric/phenolic strength of 14,000 psi. It should also increase the reliability of the turbulence ring section and the faceplate/turbulence ring joint relative to the leak paths or stresses that might be induced by axial shrinkage of the charred liner.



LONG TANK QUAL "B" CHAMBER LINER

Figure 5-30. Long Tank Qual B Chamber Liner

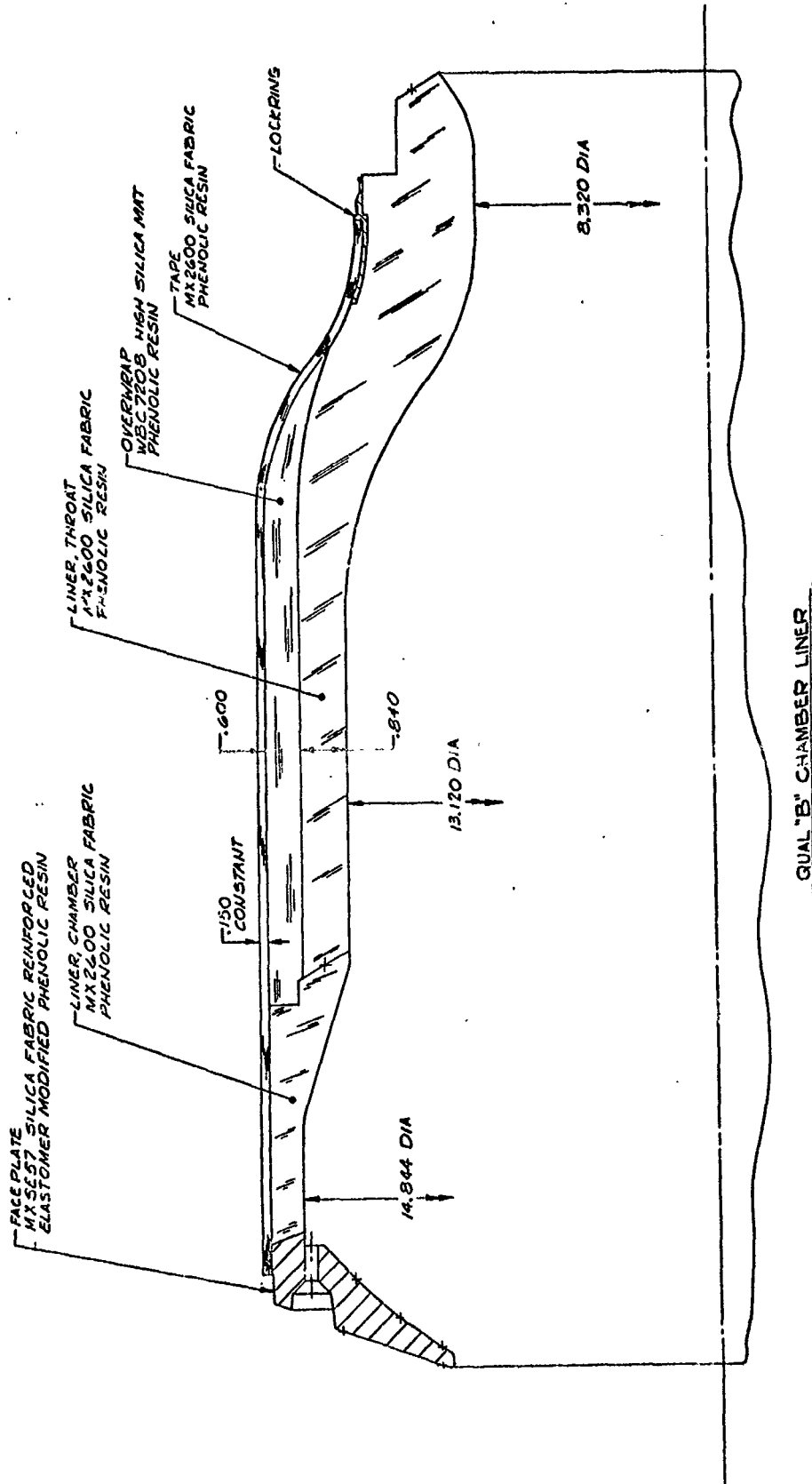


Figure 5-31. Qual B Chamber Liner

- (d) Additional lightweight backup material in the turbulence ring region and convergent section, resulting in a weight saving of about one pound by the replacement of heavyweight material in these cooler operating regions.

Processing difficulties were experienced during the liner fabrication. Initially, both the throat and turbulence ring billet exhibited delaminations as molded. The turbulence billet was rejected and a replacement part was made. This one, too, exhibited minor delaminations. Both the throat and replacement turbulence ring billet were reimpregnated with EC-201 phenolic resin and processing was resumed. After cure of the final quartz-fabric tape overwrap, two delaminations occurred in the turbulence ring, several in the cylindrical section, and two in the throat. These are shown in Figures 5-32 and 5-33. The delaminations have been accentuated for the photographs by an alcohol surface wipe.

It was believed that the defects would survive the firing. They were filled with HT-427, the epoxy-novolac adhesive used for bonding chamber liner joints. The only remaining processing was bonding of the face-plate, and finish-machining proceeded to completion.

The liner was assembled into a chamber with a titanium shell and exit cone liner transferred from the LMDE program. The chamber was assembled into Engine 1034-5 with the head end assembly borrowed from the LMDE program and previously used in a nominal extended G mission duty cycle test of the current Qual B LMDE silica chamber, Engine 1034-4. For direct comparison purposes the injector configuration was the same as that of deliverable LMDE's for the duty cycle tests. (This same head end assembly was also used for the duty cycle test of Engine 1034-6, with the quartz Qual B chamber, described in Section 5.5). In streak throat acceptance tests, the " T_{120} " time for HEA 034 was 78 seconds, indicating a high erosion rate at FTP.

5.4.2 Test Results

The extended G mission duty cycle is shown in Figure 5-34. The complete test series for the chamber, HATS 416-422, is summarized in

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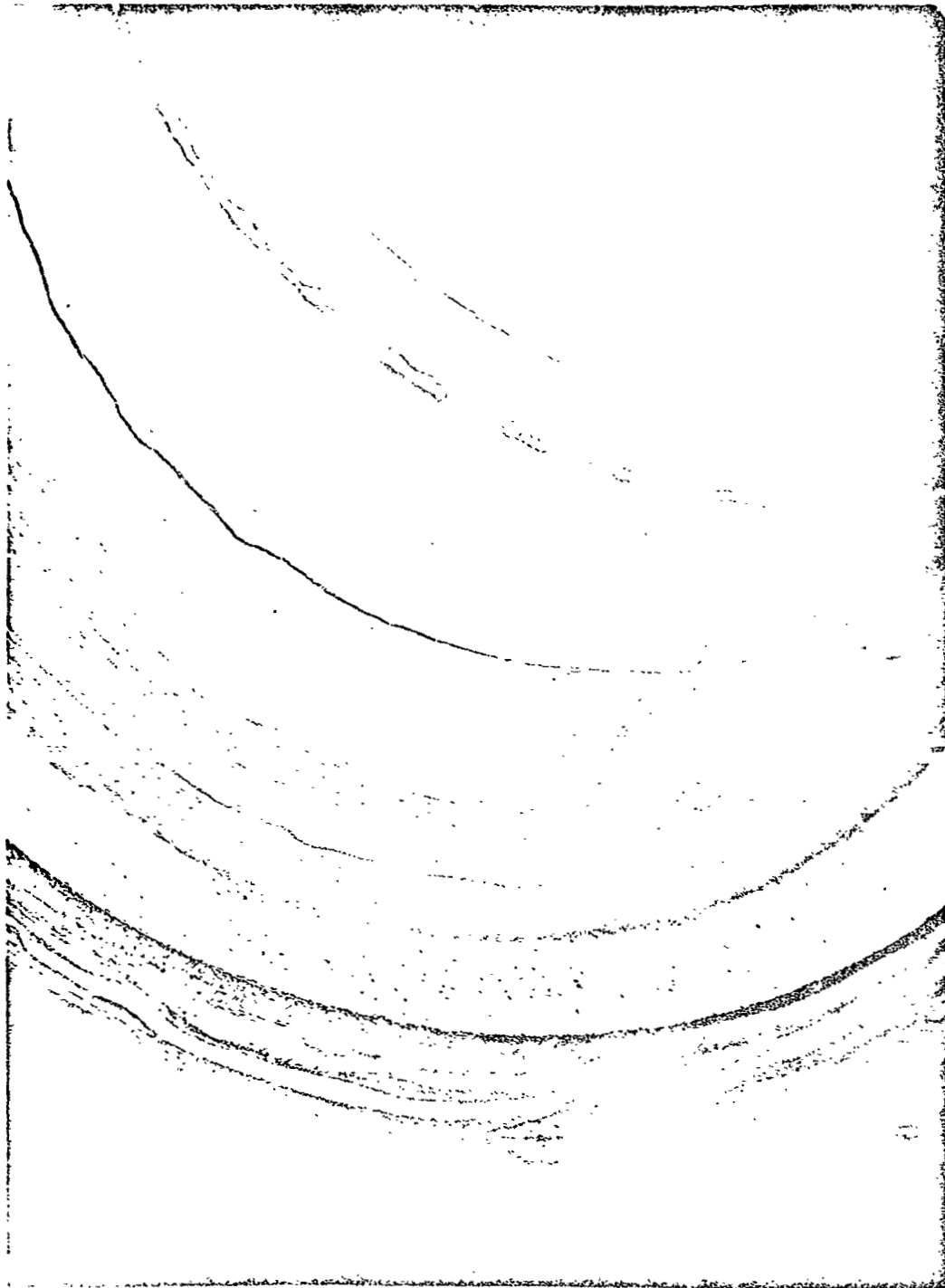


Figure 5-32. Delaminations in Prototype Quartz-Fabric/Phenolic Liner Viewed from Aft End

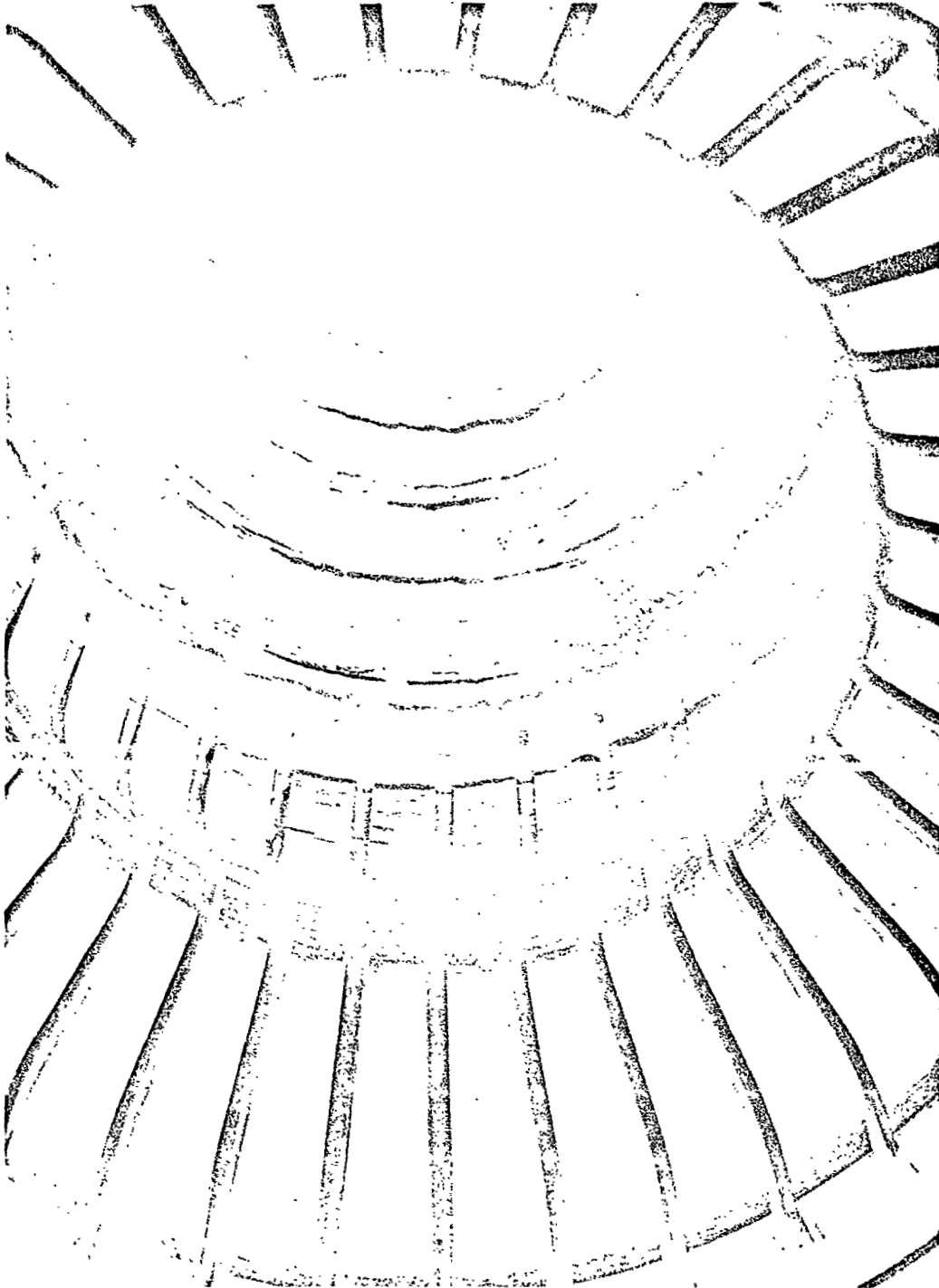
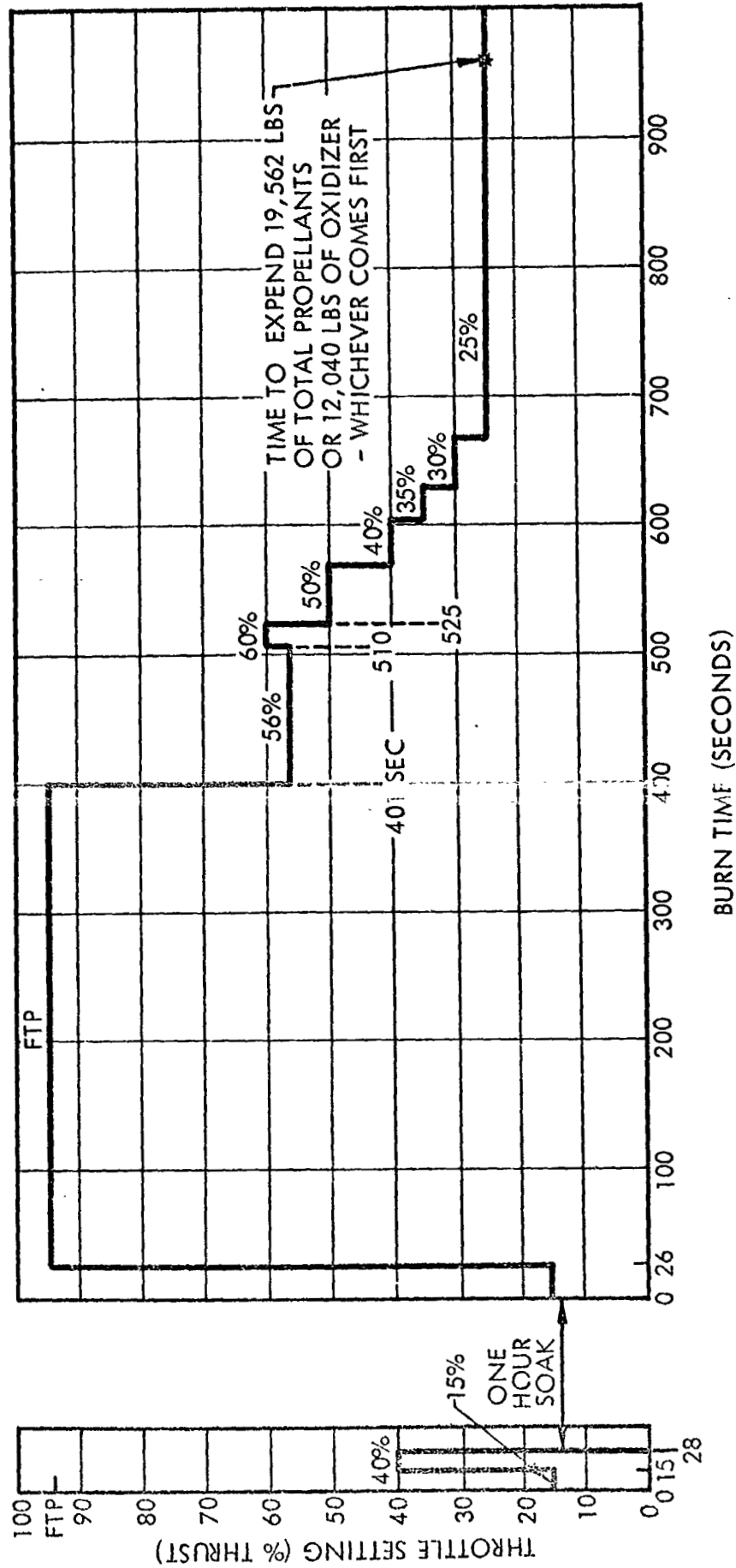


Figure 5-33 Delaminations in Prototype Quartz-Fabric/Phenolic Liner Viewed from Forward End



NOTE: ESTIMATED TOTAL TIME FOR BOTH BURNS IS 988 SECONDS

Figure 5-34. Extended G Mission Duty Cycle

Table 5-7. After HATS 416-418, to evaluate the effect of minor injector refinements (Section 3.10) the HEA was readjusted to the normal range of LMDE deliverables for completion of the test series.

HATS 419 and 420 were checkout tests of 22 and 35 seconds, respectively. HATS 421 simulated the DOI phase of the extended G mission duty cycle. HATS 422 simulated the PDI burn, requiring 985 seconds to consume 19,726 pounds of propellants. Total accumulated firing time on the chamber was 1169 seconds for HATS tests 416 through 422.

Calculated throat area increase was 14.3 percent compared to 33.2 percent for the LMDE Qual B silica-fabric/phenolic chamber with the same duty cycle. After scraping the throat, the measured throat erosion was 15.6 percent. The throat erosion profile and sections along the maximum and minimum erosion planes are presented in Figures 5-35 through 5-37.

The completely sectioned chamber is shown in Figure 5-38. Fabrication imperfections do not appear to have compromised the structural integrity of the liner. Char and erosion depth plots along the maximum and minimum erosion planes are presented in Figure 5-39. The severely eroded quartz turbulence ring is compared with its original profile in Figure 5-40.

Significant performance parameters are plotted as a function of firing time in Figure 5-41. Table 5-8 compares performance values at various throttle settings with those measured during the comparable Qual B chamber firing of Engine 1034-4. Of interest here is the fact that a measured performance advantage for Contour C appears to exist only at low thrust. At the same 25 percent FTP, however, there is a marked superiority in throat durability of quartz over silica, whereas erosion appears to be equivalent at other throttle settings. This effect is depicted graphically in Figure 5-42. Performance is discussed in detail in Reference 37.

The composite throat Contour C chamber (Section 5.3) had showed the greatest char penetration on the downstream side of the turbulence ring, about 1.5 inches farther upstream from where the same effect, and

Table 5-7

TEST SUMMARY

LONG TANK PROGRAM/CIC QUARIZ CHAMBER

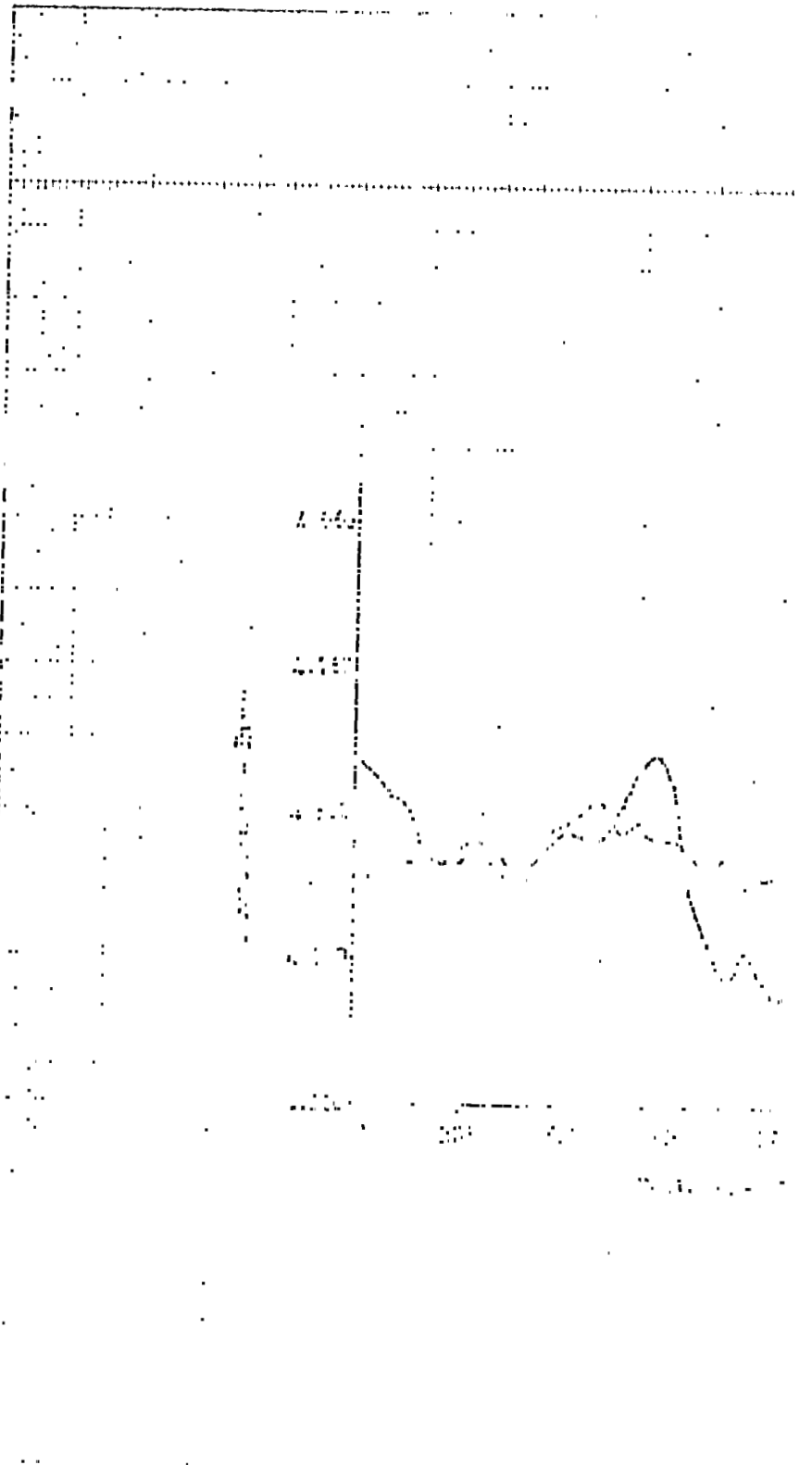
ENGINE 1034-5

TEST TYPE	TEST NO	DATE	DURATION (SEC)	ACCUMULATED DURATION (SEC)	TURBULATOR (IN)	THROTTLE RATIO	NOMINAL PROPELLANT CONDITIONING (PSIA)	REMARKS
Calibration	HATS-416	9-29-69	31	31	1.000	4.63	65 GN ₂	Calibration at TCV = 13.36 volts, 50% and 25% thrust. Throttle ratio set at 4.63. Requires fuel valve opening and injector closure for MR and A P injector adjustment.
Helium Characterization	HATS-417	9-30-69	43	74	1.000	4.63	240 Helium	Performance characterization with 1.000 inch turbulator. Shallow throttle at 13.33, 13.21 and 13.08 volts, 50% and 25%.
Helium Characterization	HATS-418	9-30-69	25	99	0.925	4.63	240 Helium	Performance characterization with 0.925 inch turbulator. Shallow throttle at 13.82 and 13.48 volts. I _{SPV} was an unacceptable low of 301.2 sec DELO = .168 at 13.82 volts
Calibration/ Acceptance	HATS-419	10-2-69	22	121	1.000	4.90	65 GN ₂	Changed throttle ratio to standard 4.90 and installed 1.000 inch turbulator. (DELO=.160) Conduct calibration test, injector too open in cav. region.
Helium Acceptance	HATS-420	10-3-69	35	156	1.000	4.90	240 Helium	Closed injector 0.003 inches. Conducted abbreviated helium baseline pre-MDC test at FTP, 50%, 25%.

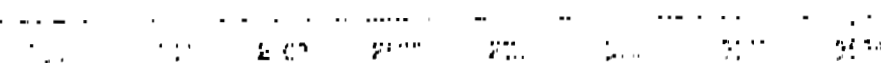
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Table 5-7 (Cont'd)
TEST SUMMARY
LONG TANK PROGRAM/CIC QUARTZ CHAMBER
ENGINE 1C34-5

TEST TYPE	TEST NO	DATE	DURATION (SEC)	ACCUMULATED DURATION (SEC)	TURBULATOR (IN)	THROTTLE RATIO	NOMINAL PROPELLANT CONDITIONING (PSIA)	REMARKS
D.O.I. DUTY CYCLE	HATS-421	10-3-69	28	184	1.000	4.90	240 Helium	First burn of duty cycle at 15% 40% followed by 1 hour soak.
P.D.I. DUTY CYCLE	HATS-422	10-3-69	985	1169	1.000	4.90	240 Helium	Second burn of duty cycle. Expended 19,726 lbs of pro- pellant at MR = 1.595 and I _{SPV} = 300.86. Combustion roughness at 56% and 60% caused loss in I _{SPV} . Erosion at 25% lower than Qual "B".



FOLDOUT FRAME /



1. *How do you feel about the way the company is doing?*

Project Name	Start Date	End Date	Status
Project A	2023-01-15	2023-03-31	Completed
Project B	2023-04-01	2023-06-30	In Progress
Project C	2023-07-01	2023-09-30	On Hold
Project D	2023-10-01	2023-12-31	Planned

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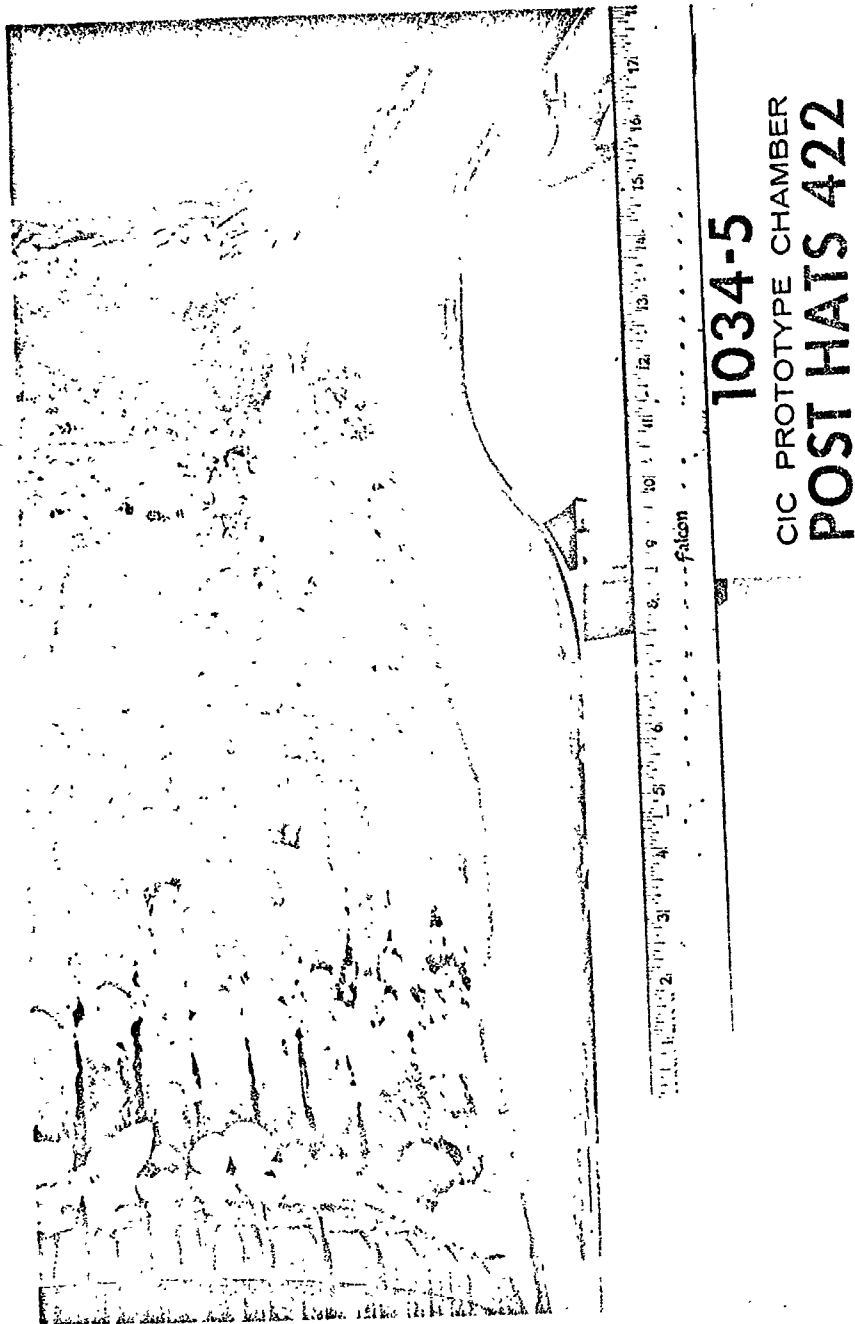


Figure 5-36. Maximum Erosion Plane

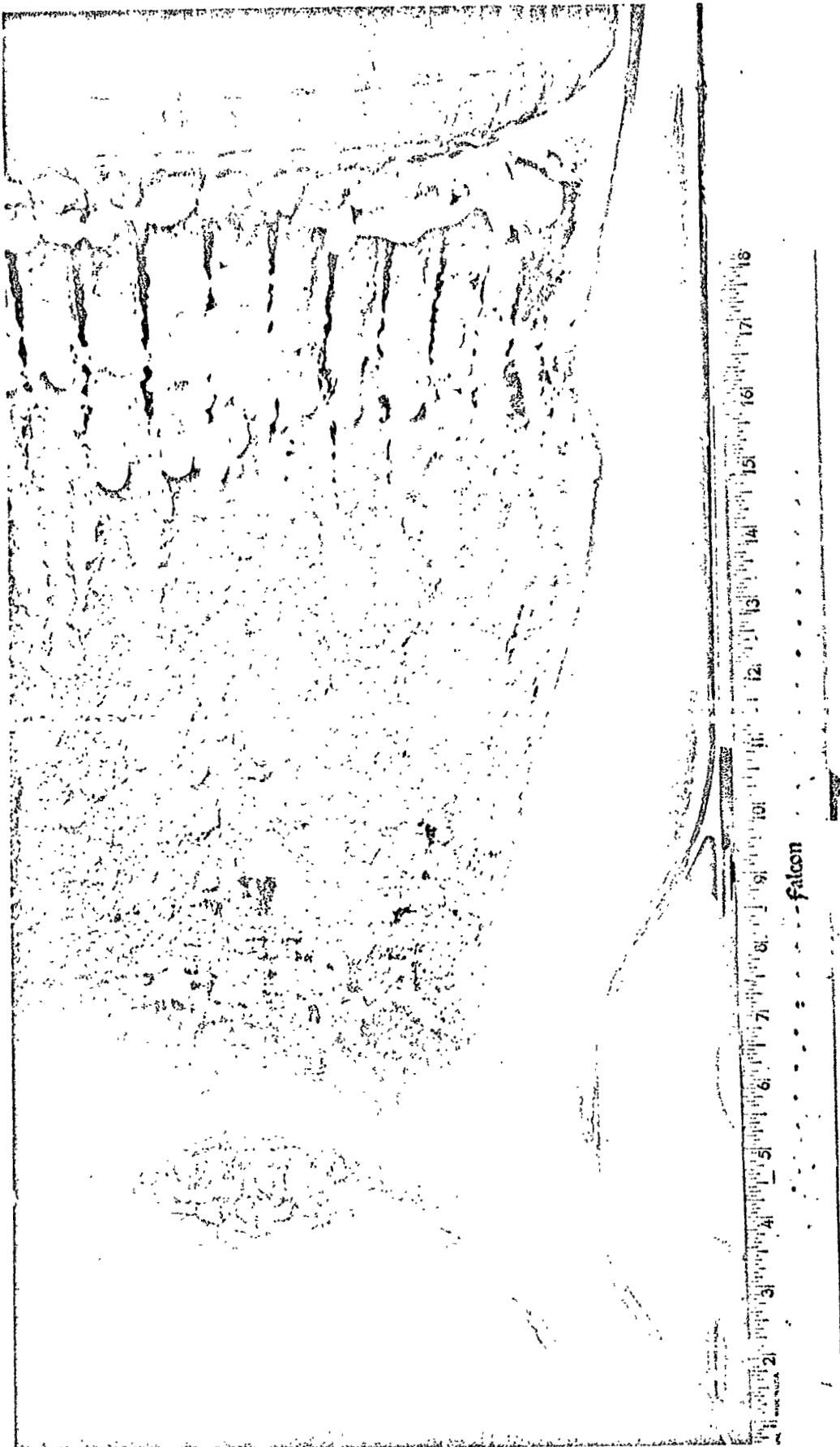


FIGURE 11

Figure 5-37. Minimum Erosion Plane

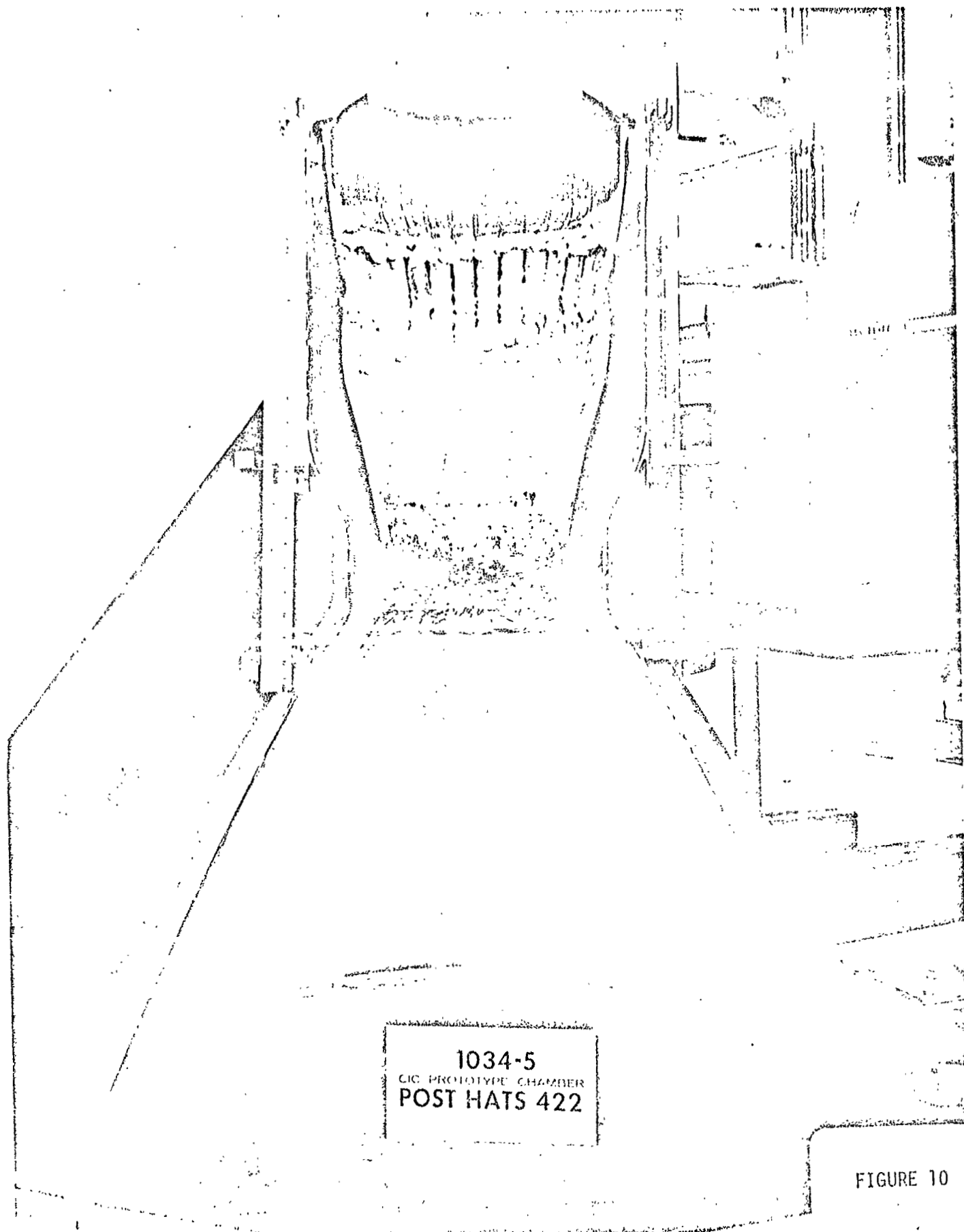
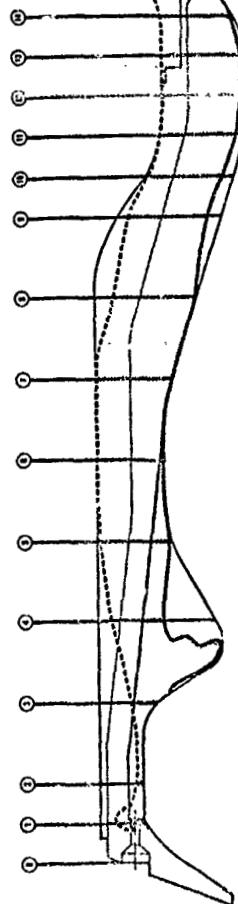


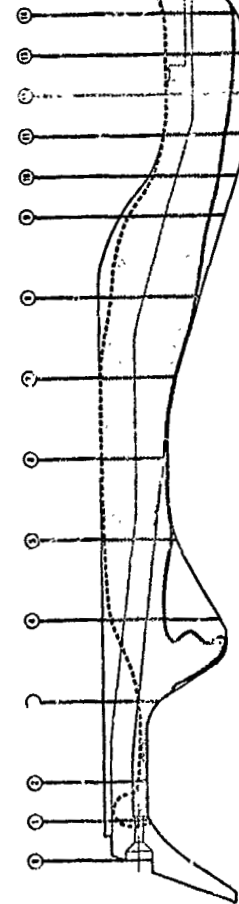
Figure 5-38. Sectioned Chamber

MINIMUM EROSION (120°)															
STATION		1	2	3	4	5	6	7	8	9	10	11	12	13	14
SLOT	TOTAL	0.80	0.80	1.00	1.21	1.38	1.64	2.08	2.72	2.28	1.96	1.93	2.02	2.00	1.80
	CHAR	0.40	0	0	0.82	1.23	1.64	2.08	2.10	2.02	1.86	1.93	2.02	2.00	1.80
TOOTH	TOTAL	1.15	1.20	1.64	1.80	2.02									
	CHAR	0.70	0.20	0.71	1.44	1.84									



MAXIMUM EROSION (0°)

STATION	1	2	3	4	5	6	7	8	9	10	11	12	13	14	
SLOT	TOTAL	0.80	0.80	1.21	1.26	1.50	1.54	1.89	2.44	2.26	1.78	1.54	1.64	1.72	1.78
	CHAR	0.46	0	0	1.00	1.44	1.54	1.89	2.16	2.06	1.68	1.54	1.64	1.72	1.78
TOOTH	TOTAL	1.15	1.18	1.72	1.70	1.87									
	CHAR	0.80	0.28	0.78	1.44	1.78									



1034-S CC PROTOTYPE CH - 234

Figure 5-39. Char and Erosion Pattern

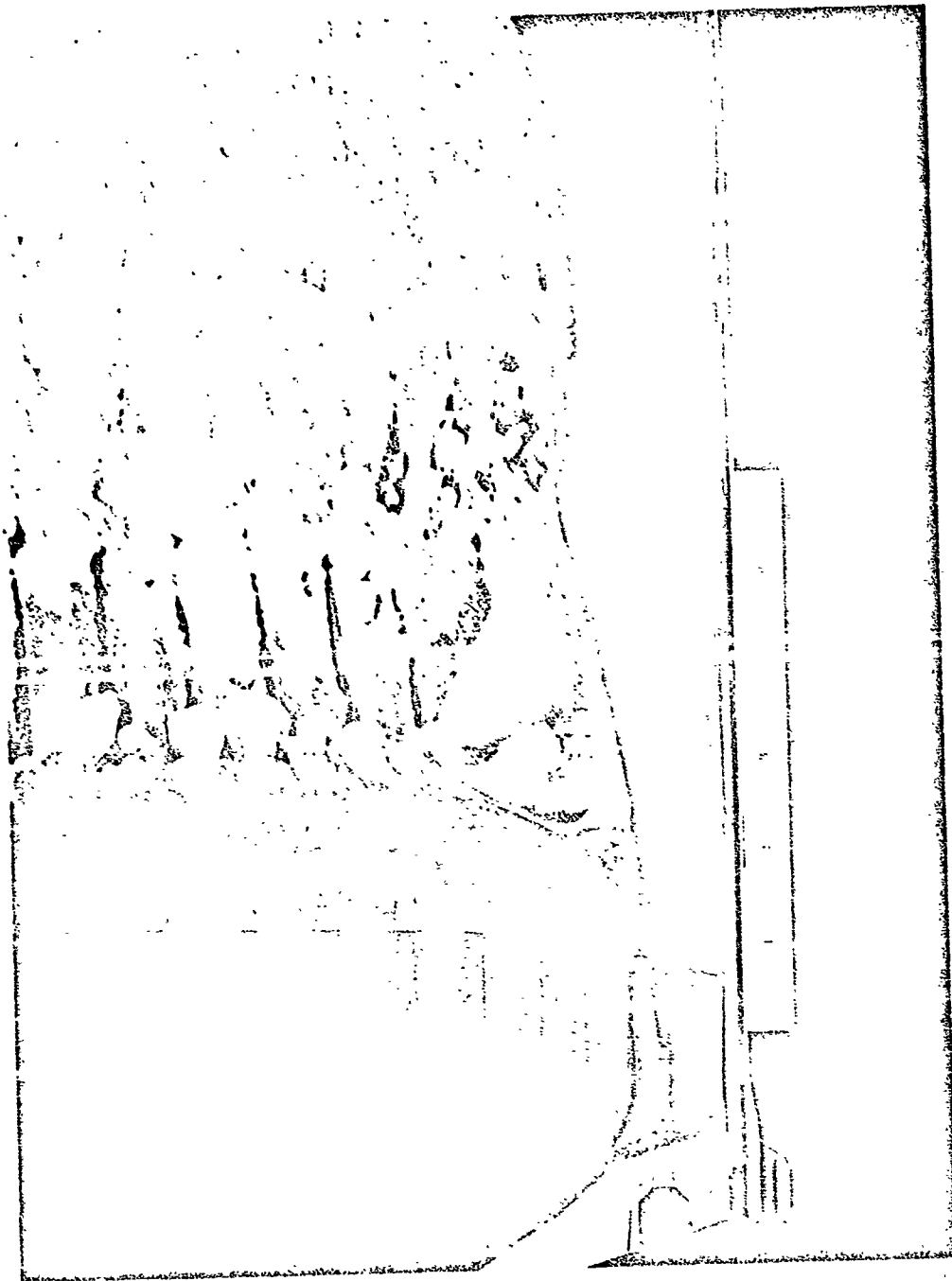
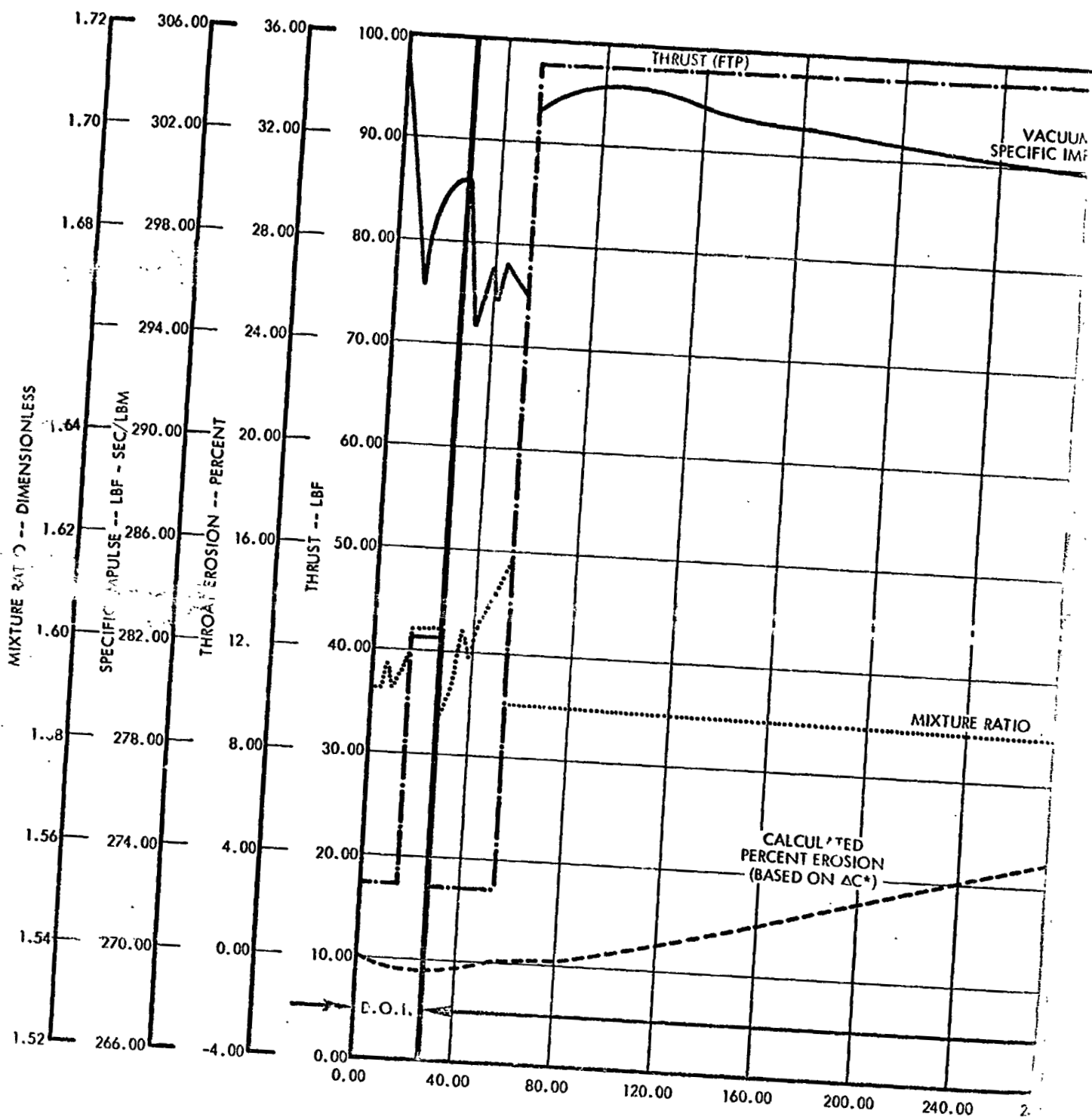
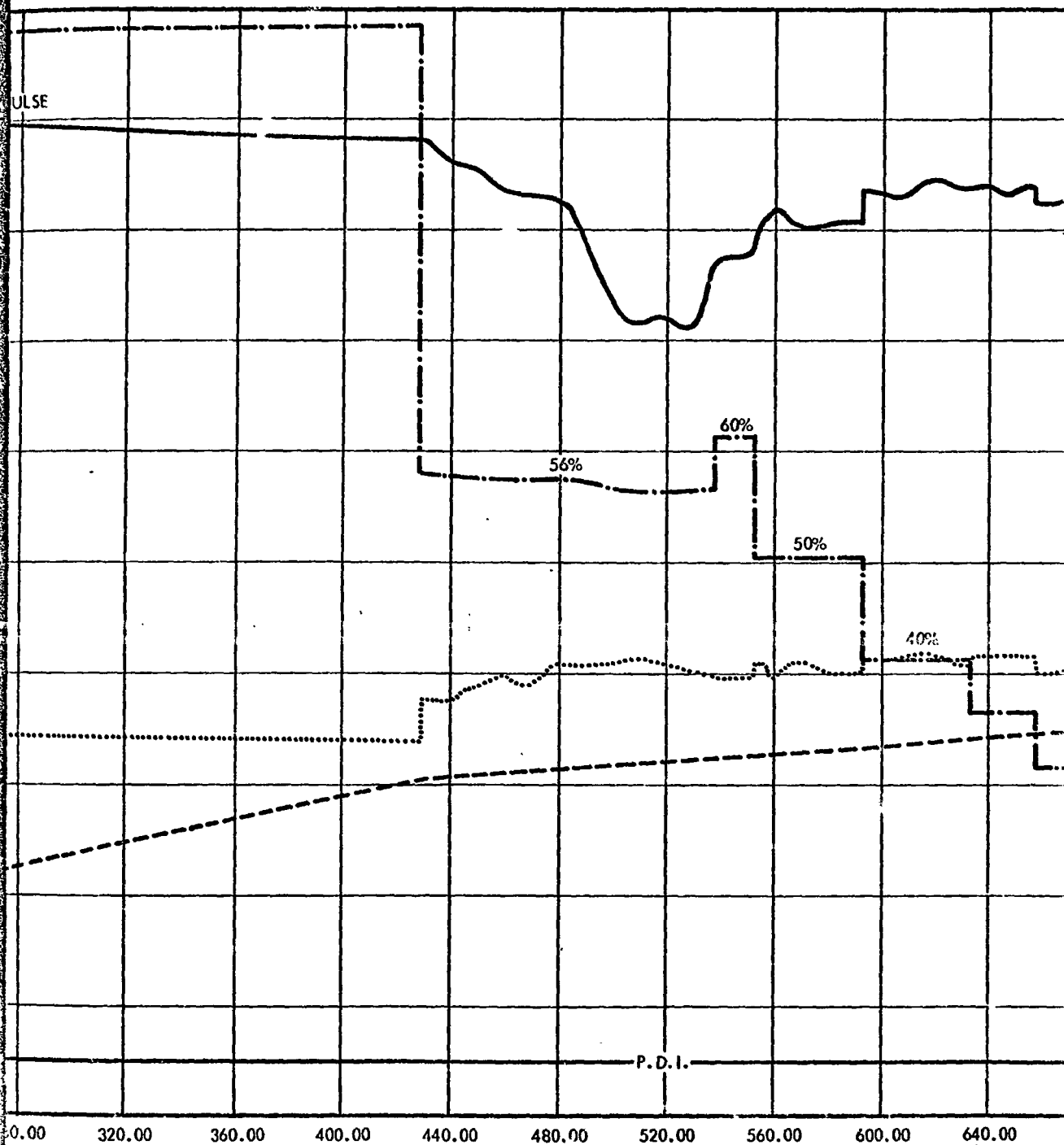


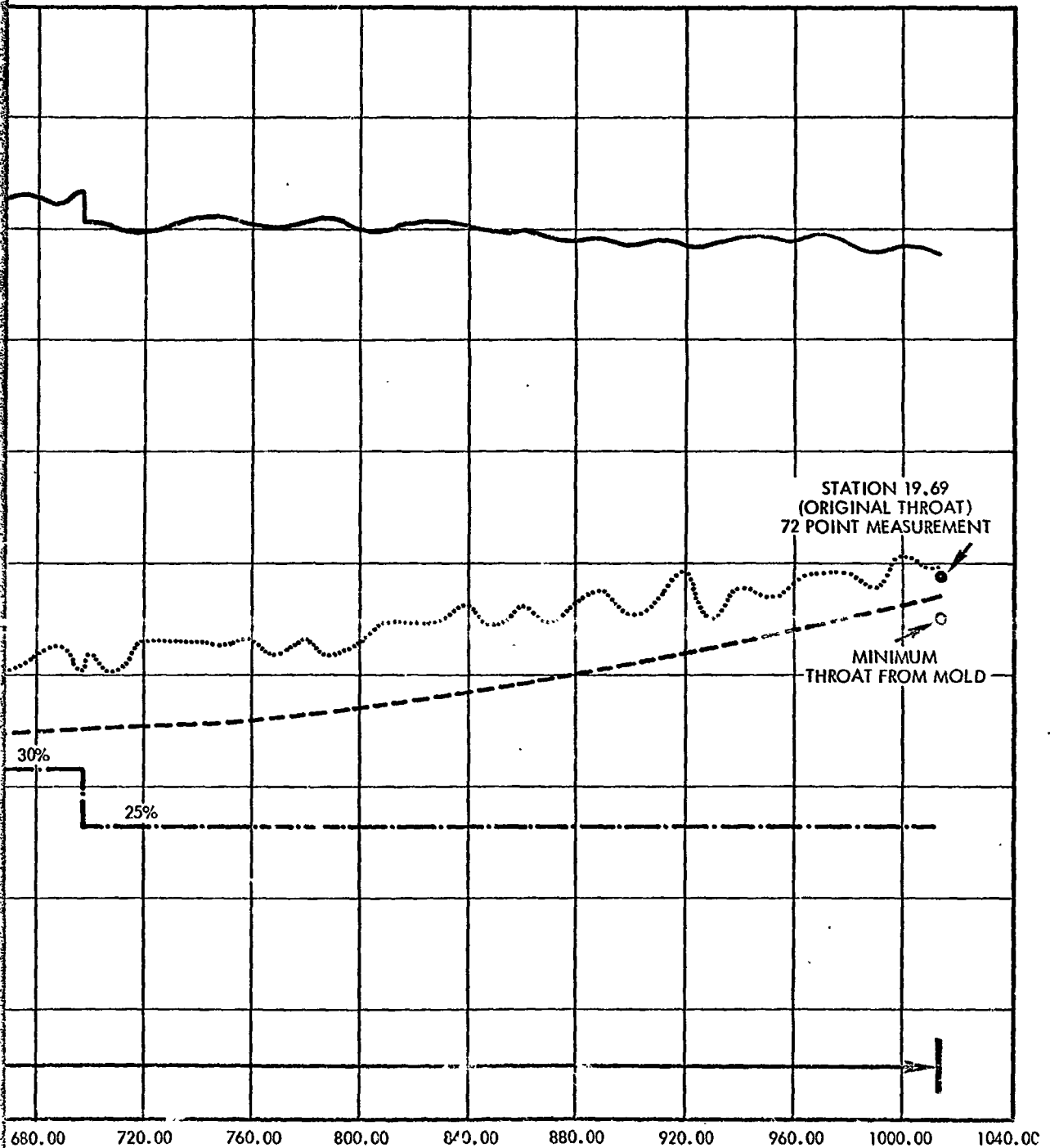
Figure 5-40. Prototype Chamber Turbulence
Ring Erosion



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FOLDOUT FRAME 2



FOLDOUT FRAME 3

Figure 5-41. Prototype Chamber Performance, Engine 1034-5

TABLE 5-8

LONG TANK PROGRAM

EXTENDED MDC - NOMINAL CONDITIONS

QUAL "B" SILICA CHAMBER	ENGINE 1034-4
HATS-391 and 392	
<u>CIC QUARTZ CHAMBER</u>	ENGINE 1034-5
HATS-421 and 422	

	MDC PHASES				AVERAGE DOI & PDI TO OX. DEPLETION	MDC PHASES				AVERAGE DOI & PDI TO TOTAL PROPELLANT DEPLETION
	FTP	56%	56-25%	25%		FTP	56%	56-25%	25%	
I _{SPV} (Avg)	302.3	296.0	297.7	292.7	299.7	302.4	297.7	298.7	297.9	300.8
Mixture Ratio	1.595	1.620	1.620	1.635	1.607	1.589	1.600	1.601	1.611	1.595
Erosion-% (Calculated)	8.5*	0.2	1.1	21.6	30.1	8.2*	← 2.0 →		4.6	14.4

***Starts at -1.6% Erosion**

***Starts at -0.5% Erosion**

Time to oxidizer depletion (DOI & PDI) = 995 sec.
Time to propellant depletion (EOL & PDI) = 998 sec.
Propellant depletion at 19,562 lbs total.
Time to end of test = 1029 sec.
% erosion at end of test 32.3%.

Time to propellant depletion (DOI & PDI) = 994 sec.
 Propellant depletion of 19,562 lbs total.
 Time to end of test = 1013 sec.
 % erosion at end of test = 14.8%.

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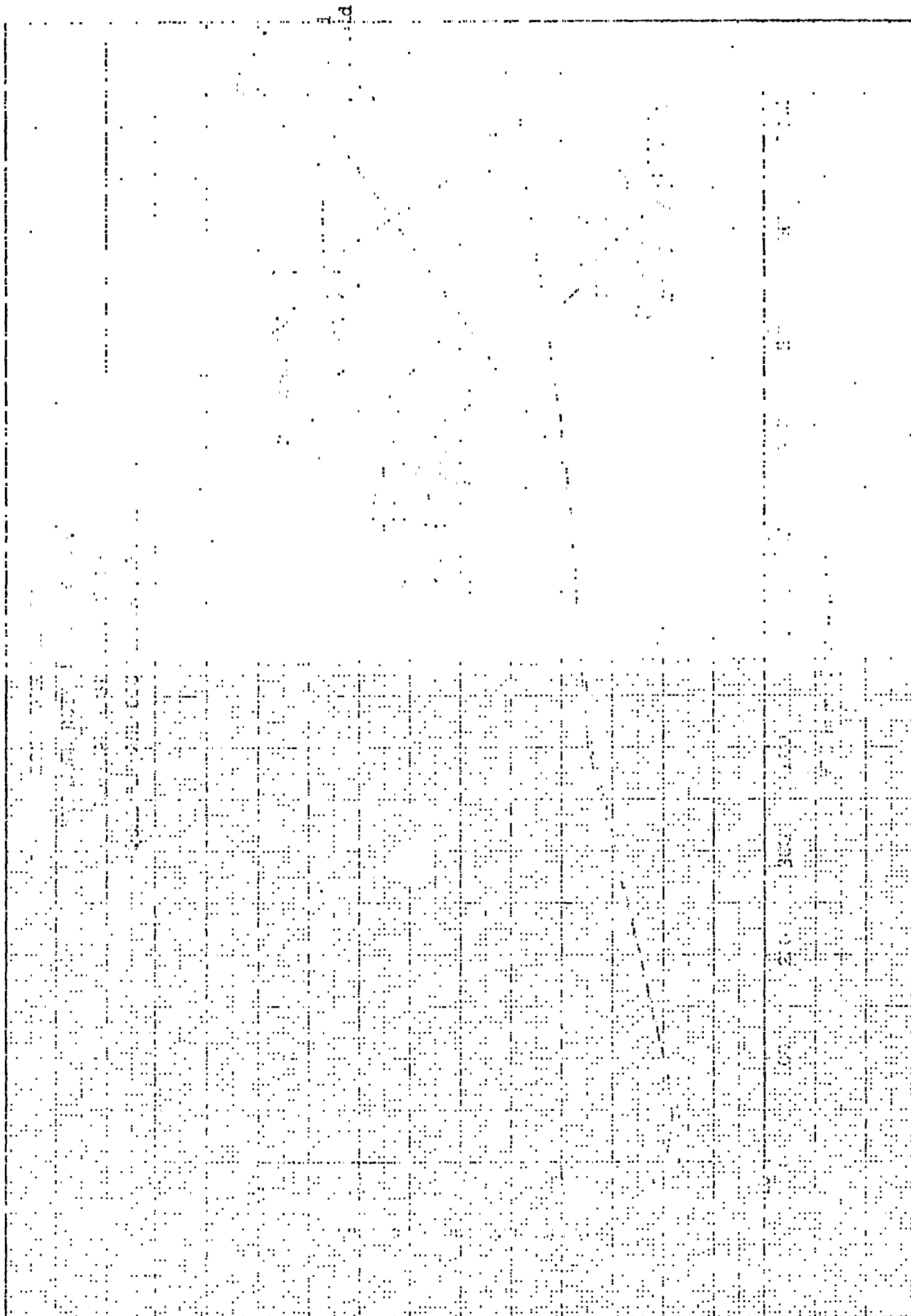


Figure 6

Figure 5-42. Quartz vs. Silica Erosion

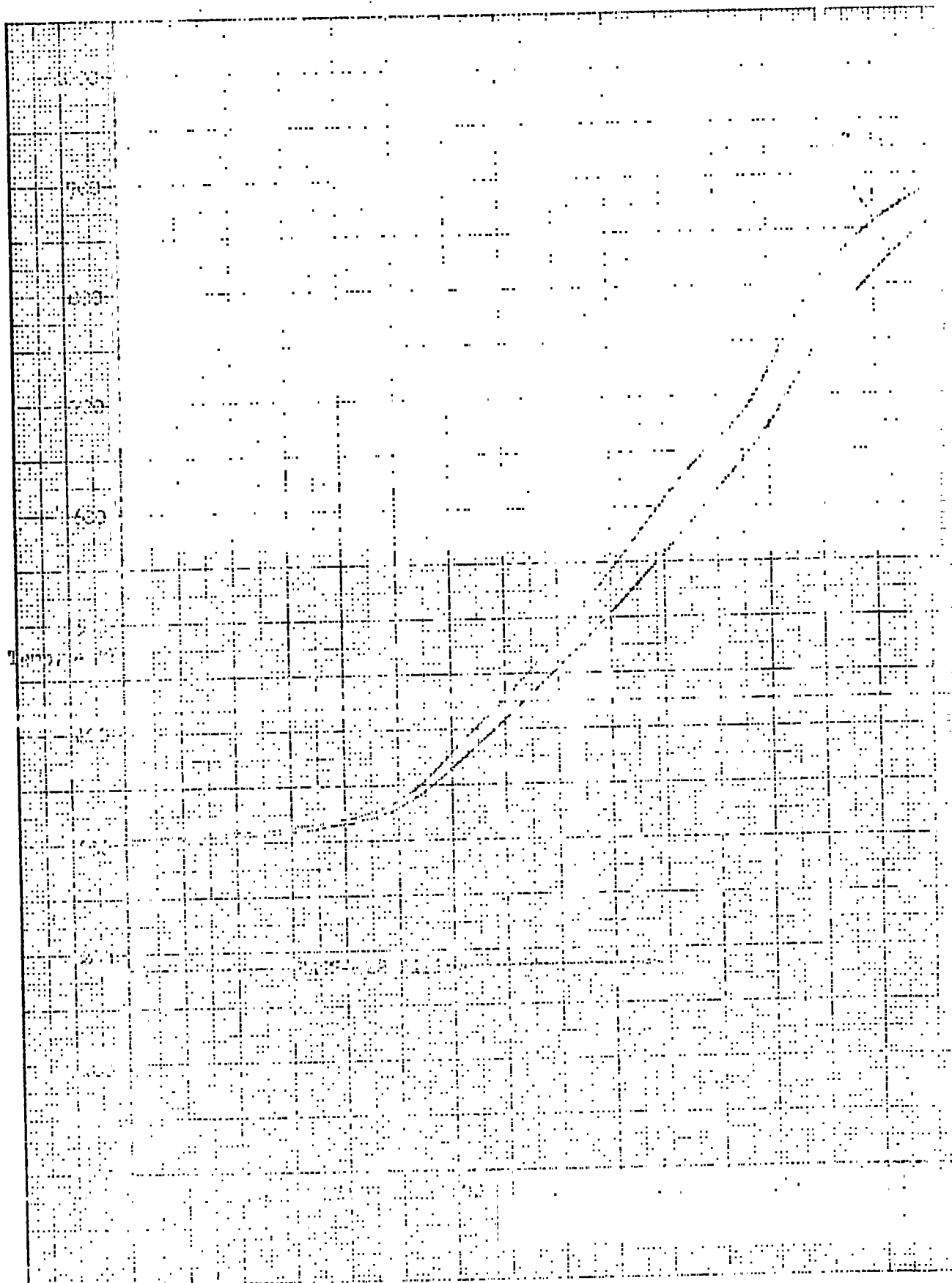
highest titanium case backwall temperatures, are usually observed on the Qual B chamber. The highest shutdown temperature measured at this station, 10 inches from the head end attachment flange, was 595°F, as shown in Figure 5-43. This value was near the low end of the range measured for the Qual B chamber. Figure 5-44 compares the temperature data and indicates that the Qual B backwall temperature was rising at a much greater rate at shutdown.

5.4.3 Conclusions

- Except at low thrust, the Contour C measured specific impulse over the throttle range offered no marked improvement over that of the current Qual B contour.
- The quartz laminate offers a significant improvement in erosion resistance over its silica counterpart. This reduced erosion was experienced despite the existence of delaminations and loss of orientation in the liner as fabricated.
- Erosion on the downstream side of the slotted quartz-fabric/phenolic turbulence ring was just as severe as that experienced previously with the silica-fabric/phenolic Contour C turbulence ring of the Advanced Materials Test Chamber (Section 5.3).
- The intermediate quartz wrap performed well as a secondary gas seal and improved the integrity of the chamber/exit cone joint.
- Titanium backwall temperatures at shutdown were at the minimum expected. At the thinnest region of the chamber, Station 21.0, the maximum shutdown temperature was only 595°F.

5.5 QUARTZ QUALIFICATION B CHAMBER

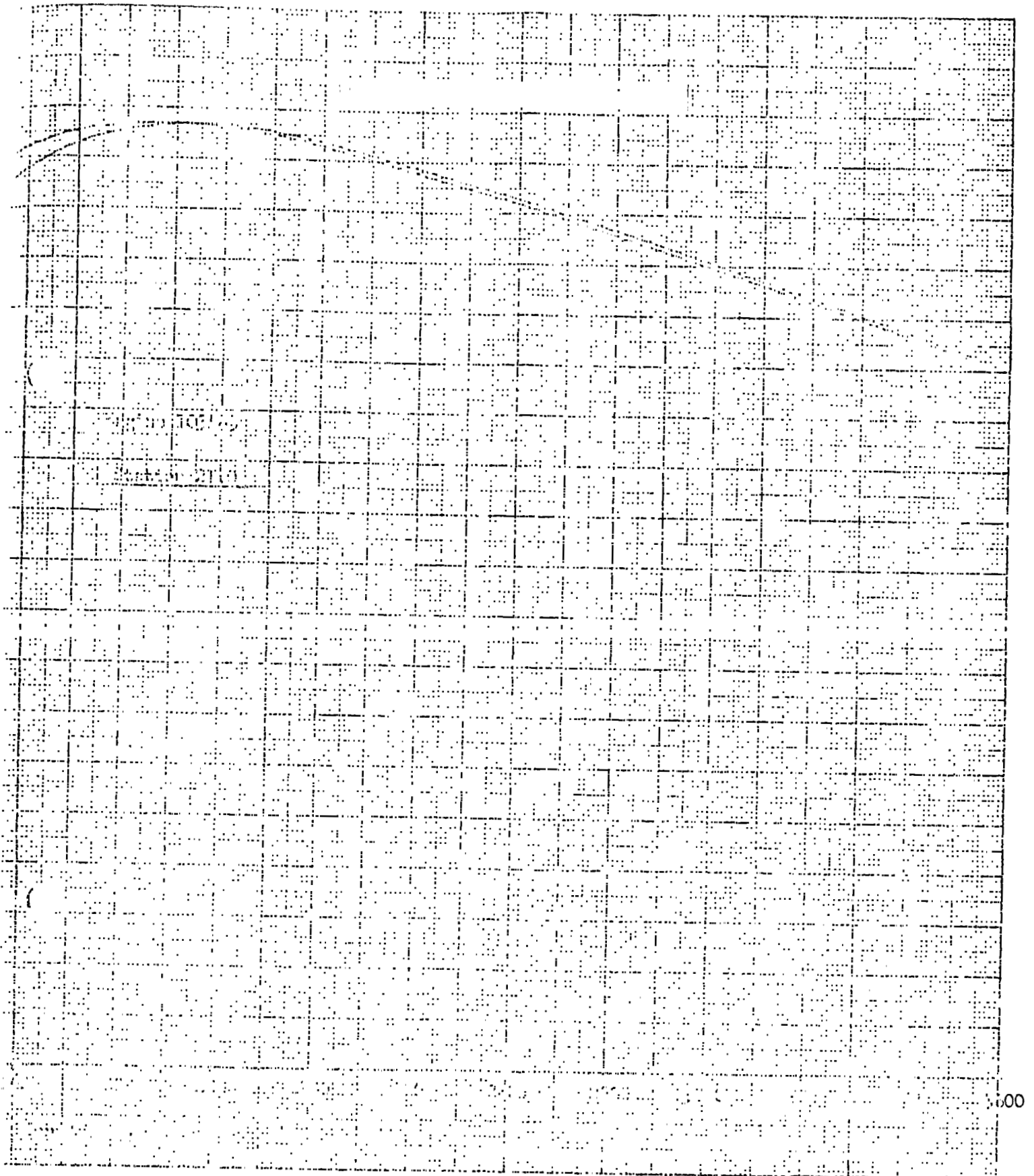
As mentioned in Section 5.4, based on the successful results of the CIC program, it was decided to evaluate a quartz chamber in the mainline



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Figure 5-43. Titanium Case Temperature

5-64

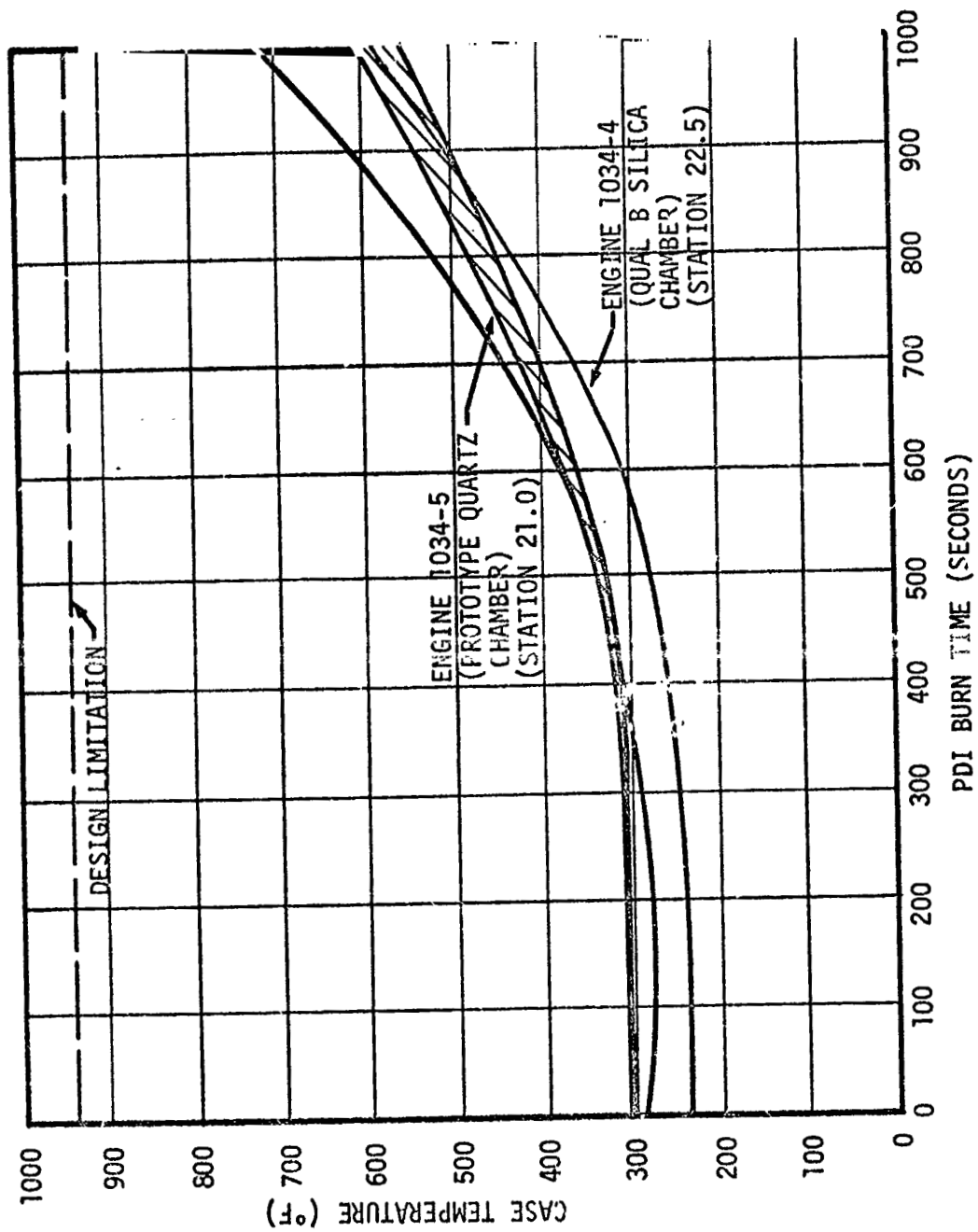


Figure 5-44. Long Tank Program Case Temperature Measurements

LMDE program to accommodate the increased propellant load for the extended G mission in a reliable manner.

5.5.1 Fabrication

A quartz chamber liner of the Figure 5-30 design was fabricated. It exhibited surface delaminations similar to those of the Prototype Contour C Chamber. Imperfections of this type, shown in Figure 5-45 and 5-46, are not unusual during process development. These were reimpregnated and the liner was eventually integrated into a Qual B assembly, designated Engine 1034-6.

An interesting feature of the design, Figure 5-30, is that it contains 0.1-inch less ablative material in the cylindrical section than that of the Qual B design, Figure 5-31. This fact, plus the fact that it was possible to include additional backup lightweight insulation accounts for the fact that the new design results in a nominal weight saving of 1.1 lb.

5.5.2 Test Results

After a checkout test, HATS 426, testing continued through HATS 432 for a total of 1,190 seconds. The Phase II portion of the duty cycle was interrupted when HATS 431 was shut down prematurely. The test was then completed with HATS 432.

HATS acceptance runs (HATS 427 and 428) were conducted with the engine adjusted to nominal conditions and performance parameters (I_{sp} , MR, and ΔP 's) were near nominal for both runs.

The extended G mission duty cycle was initiated on 16 October 1969 with a 28 second DOI burn, HATS 430, and an initial PDI burn, HATS 431. During the PDI burn, a premature engine shutdown signal was received 320 seconds into the scheduled 375 second FTP portion of the PDI burn. The premature shutdown was caused by a special facility cable shorting out and causing the shutoff valves to close.

The FTP shutdown was rough and the chamber sustained numerous pressure spikes ranging from 40 to 130 psi peak-to-peak. Inspection of the injector element showed the elements to be in good condition and acceptable

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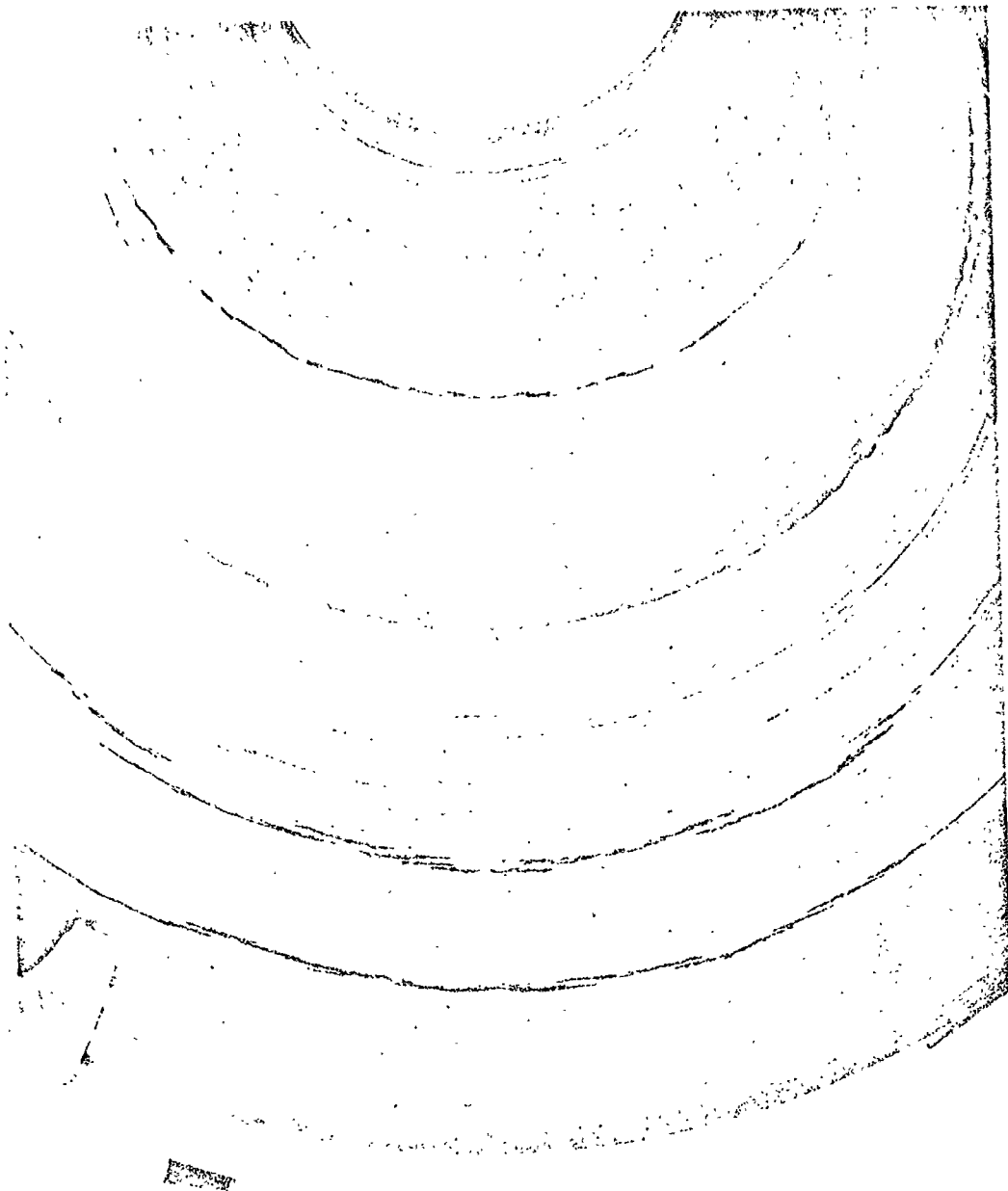


Figure 5-45. Turbulence Ring Delaminations
in Quartz Qual B Chamber Liner.

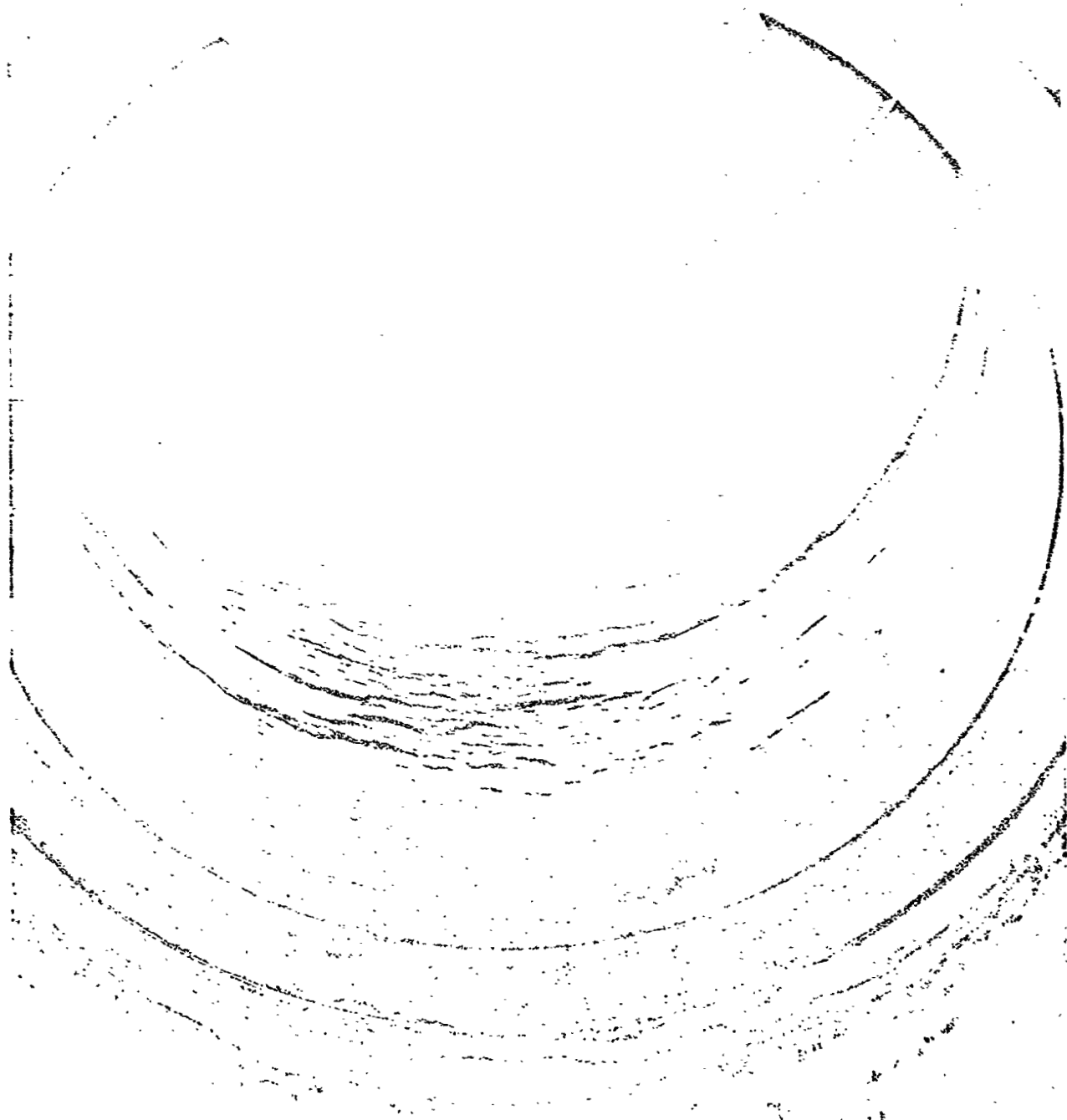


Figure 5-46. Throat Delaminations in Quartz
Qual B Chamber Liner

for further testing. Inspection of the quartz chamber revealed some delaminations in the cylindrical section of the chamber and a partial opening of the joint between the chamber liner and exit cone was subsequently filled with zinc chromate putty.

When testing was continued with HATS 432, a slight change in the thrust profile was made to allow for a warm-up phase consisting of 33 seconds at low chamber pressure so that some glass flow could occur in the chamber before throttling to FTP. A fifty-five second burn at FTP was then made which completed the scheduled 375 second FTP burn time and the balance of the profile for an extended G mission profile was then completed. Figure 5-47 is a plot of the throttle profiles for the three runs (HATS 430, 431, and 432) that constituted the extended G mission duty cycle. Table 5-9 summarizes the test series.

A total throat erosion of 7.7 percent for the total duty cycle was calculated based on c^* data. Throat erosion of 2.9 percent was experienced for the FTP portion of the burn and 4.6 percent for the 25 percent thrust portion of the duty cycle. The measured throat area change after cool down was 8.5 percent. The total burn time to deplete 12,040 lbs of oxidizer and a total propellant load of 19,562 lbs was 998 seconds. The burn times to propellant depletion exclude the 33 second warm up run that preceded the PDI continuation on HATS 432.

Performance parameters of Table 5-10 and Figure 5-48 show that specific impulse over the throttle range was equivalent to that experienced with the 1034-4 Qual B chamber (Table 5-8). Throat erosion at full thrust was less than half that of the Qual B and Prototype chambers and confirms the results for quartz-fabric/phenolic previously observed in the Advanced Materials Composite Throat Test (Section 5.3, Table 5-5). The real advantage of quartz over silica is evident at the low thrust (and reduced heat flux) level where its higher melt viscosity resists the shear removal forces.

The uniform throat erosion is shown in Figures 5-49 and 5-50. The erosion profile, Figure 5-51, shows a significant improvement over that of the Qual B silica chamber.

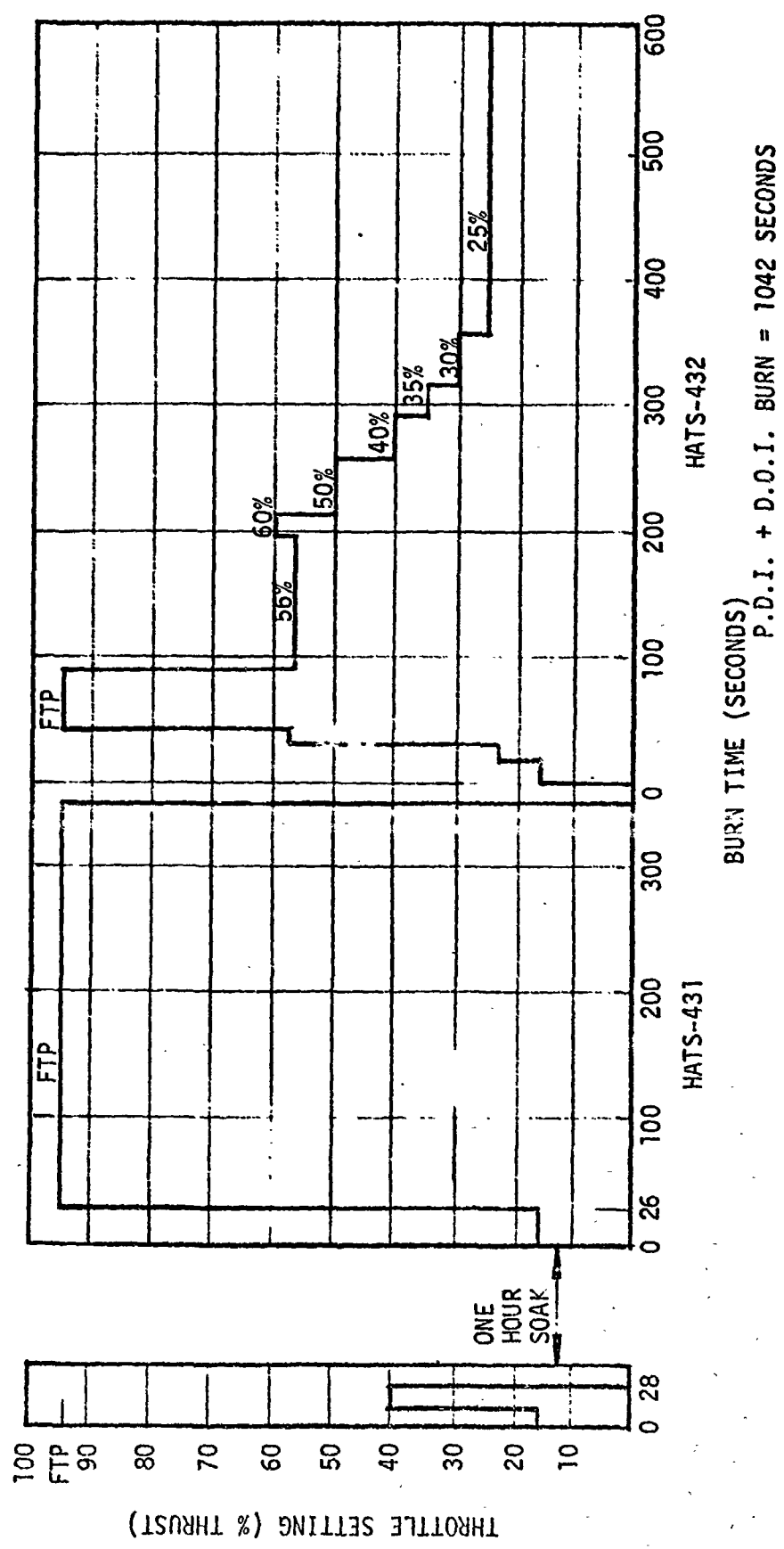


Figure 5-47. Interrupted Extended G Mission Duty Cycle

TABLE 5-9. TEST SUMMARY
LONG TANK PROGRAM/QUAL B QUARTZ CHAMBER
ENGINE 1034-6

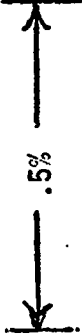
TEST NO.	TCV	%F	DURATION (SEC)	REMARKS
HATS-426	3.22	15	3	Calibration Test. 65 psia GN ₂ Saturated Propellants.
	13.80	FTP	7	
	7.50	50	5	
	4.44	25	7	
			<u>22</u>	
HATS-427	3.22	15	5	Acceptance Test. 65 psia GN ₂ Saturated Propellants.
	13.80	FTP	20	
	9.50	65	5	
	7.50	50	5	
	4.44	25	<u>5</u>	
			40	
HATS-428	3.22	15	3	Acceptance Test. 65 psia GN ₂ Saturated Propellants.
	13.80	FTP	20	
	9.00	60	5	
	7.50	50	5	
	6.28	40	5	
	4.44	25	<u>5</u>	
			43	
HATS-429	3.22	15	3	Helium Baseline. 240 psia Helium Saturated Propellants. After the checkout, acceptance, and helium baseline tests, the engine was to perform an extended "G" mission duty cycle similar to the baseline duty cycle, but because of a facility malfunction the duty cycle was per- formed with the following profile:
	13.80	FTP	20	
	9.00	60	5	
	7.50	50	5	
	6.28	40	5	
	4.44	25	<u>5</u>	
			43	
HATS-430	3.22	15	15	First burn. 240 psia Helium Saturated Propellants.
	6.28	40	<u>13</u>	
			28	
HATS-431	3.22	15	26	PDI Burn. Second burn. 240 psia Helium Saturated Propellants. An unscheduled shutdown of the engine 346 seconds into the PDI burn was due to a short in a special facility shutoff valve cable. Fuel and oxidizer purges were turned on 15 seconds after shutdown.
	13.80	FTP	<u>320</u>	
			346	

TABLE 5-9. TEST SUMMARY (Continued)

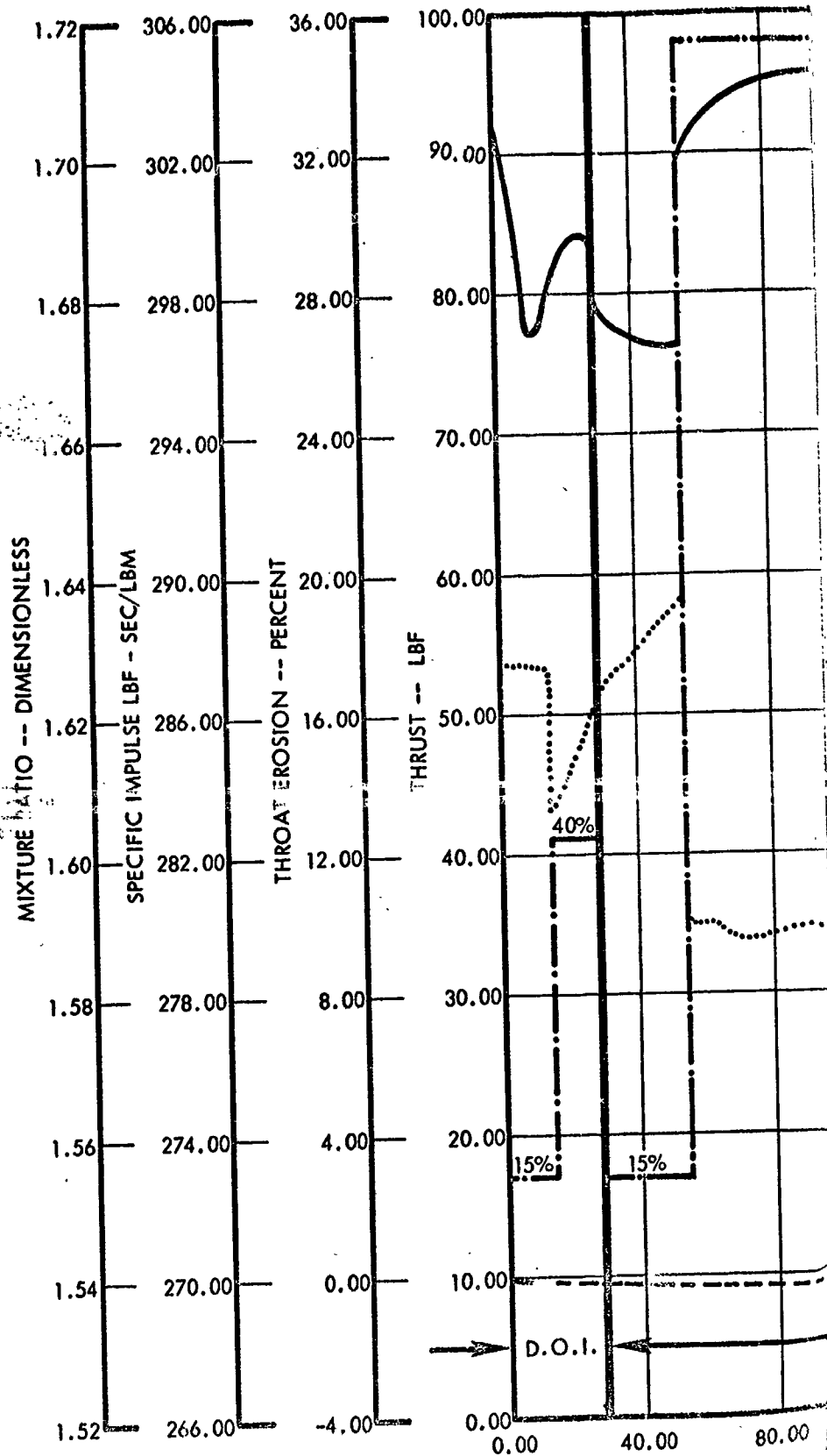
<u>TEST NO.</u>	<u>TCV</u>	<u>%F</u>	<u>DURATION (SEC)</u>	<u>REMARKS</u>
HATS-432	3.22	15	16	The first three throttle points are warm-up.
	4.44	25	10	The 55 seconds at FTP make the total FTP time equal to 375 seconds which was the original duty cycle burn time at FTP.
	8.40	56	7	
	13.80	FTP	55	
	8.40	56	109	240 psia helium saturated propellants.
	9.00	60	15	
	7.51	50	40	Fuel and oxidizer purges were turned on 15 seconds after shutdown.
	6.28	40	40	
	5.67	35	25	The accumulated firing duration on the combustion chamber was 1190 seconds.
	5.06	30	40	
	4.44	25	<u>311</u>	
			668	

TABLE 5-10
LONG TANK PROGRAM
ENGINE 1034-6 NOMINAL CONDITIONS
(QUAL "B" QUARTZ CHAMBER)

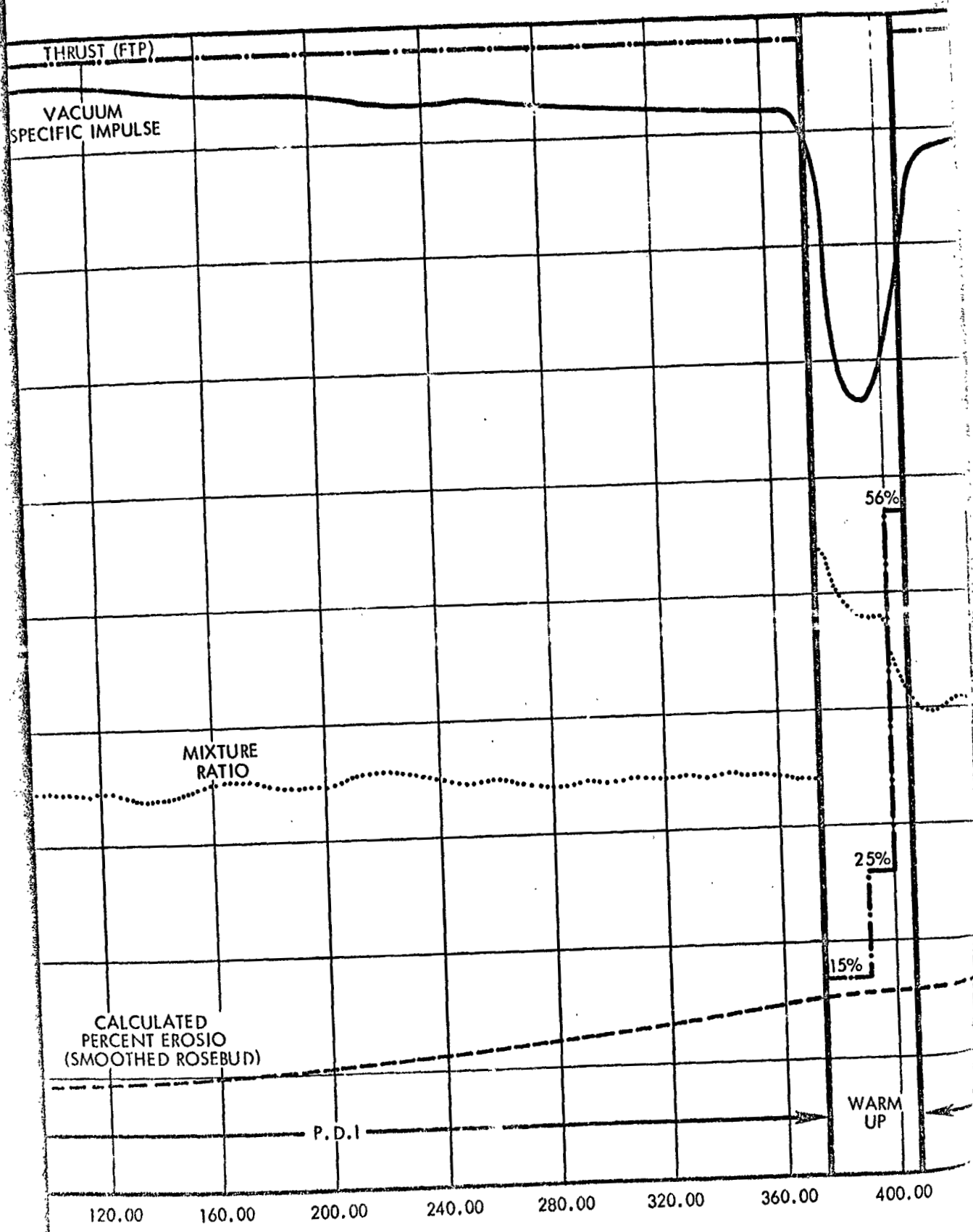
SUMMARY OF CUMULATIVE
PERFORMANCE PARAMETERS
HATS-430, 431 and 432

	<u>FTP</u>	<u>56%</u>	<u>55-25%</u>	<u>25%</u>	<u>Integrated Average DOI + PDI *</u>
Isp (Avg)	303.3	297	294.8	293.3	300.14
(LBF-Sec/LBM)					
Mixture Ratio O/F	1.591	1.613	1.616	1.627	1.602
Calculated Erosion	2.9 %				7.7%
				4.3%	TOTAL

*Minus 33 Second Warmup on HATS-432



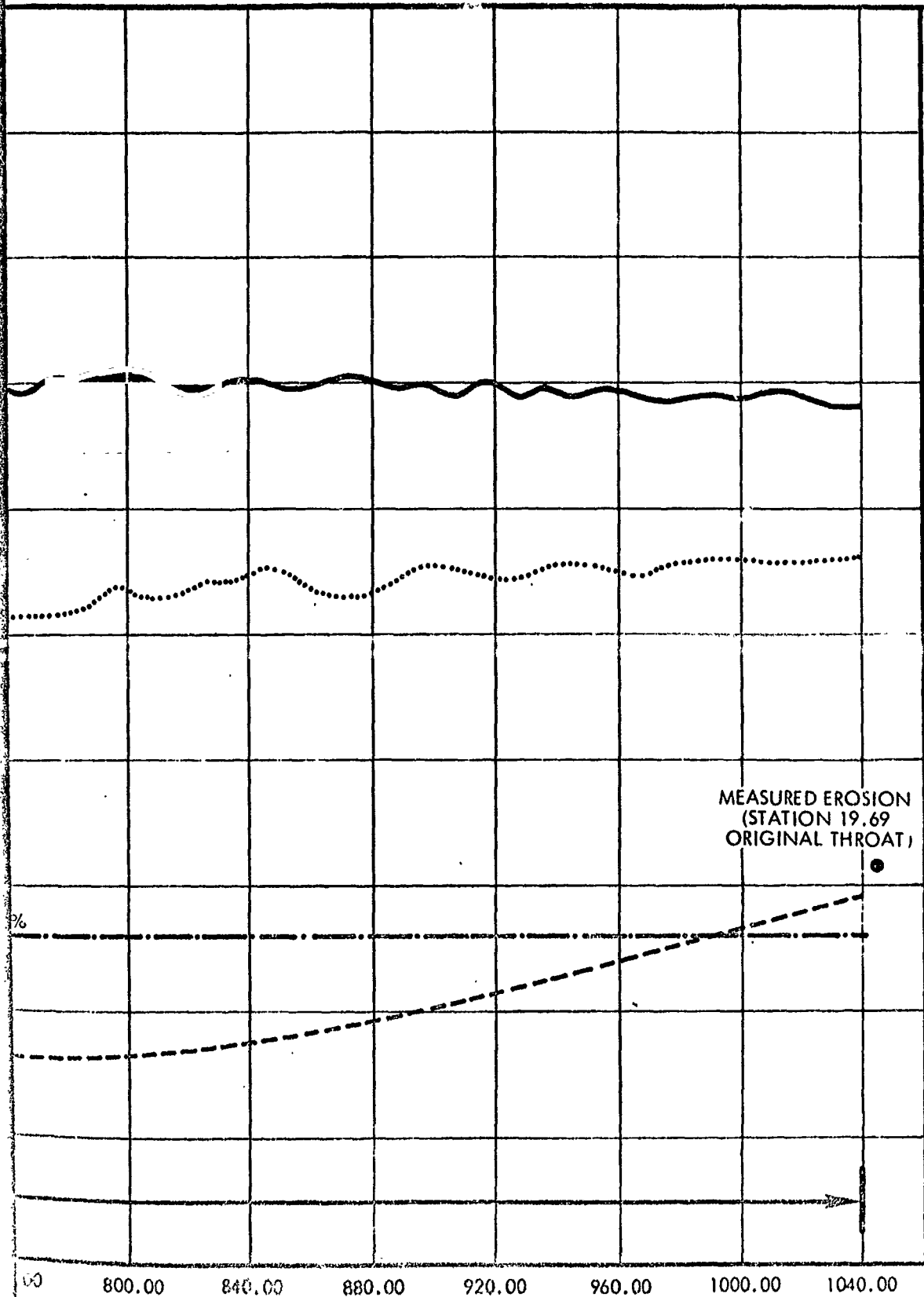
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FOLDOUT FRAME 3

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FOLDOUT FRAME 4 Figure 5-48. Qual B Quartz Chamber Performance, Engine 1034-6

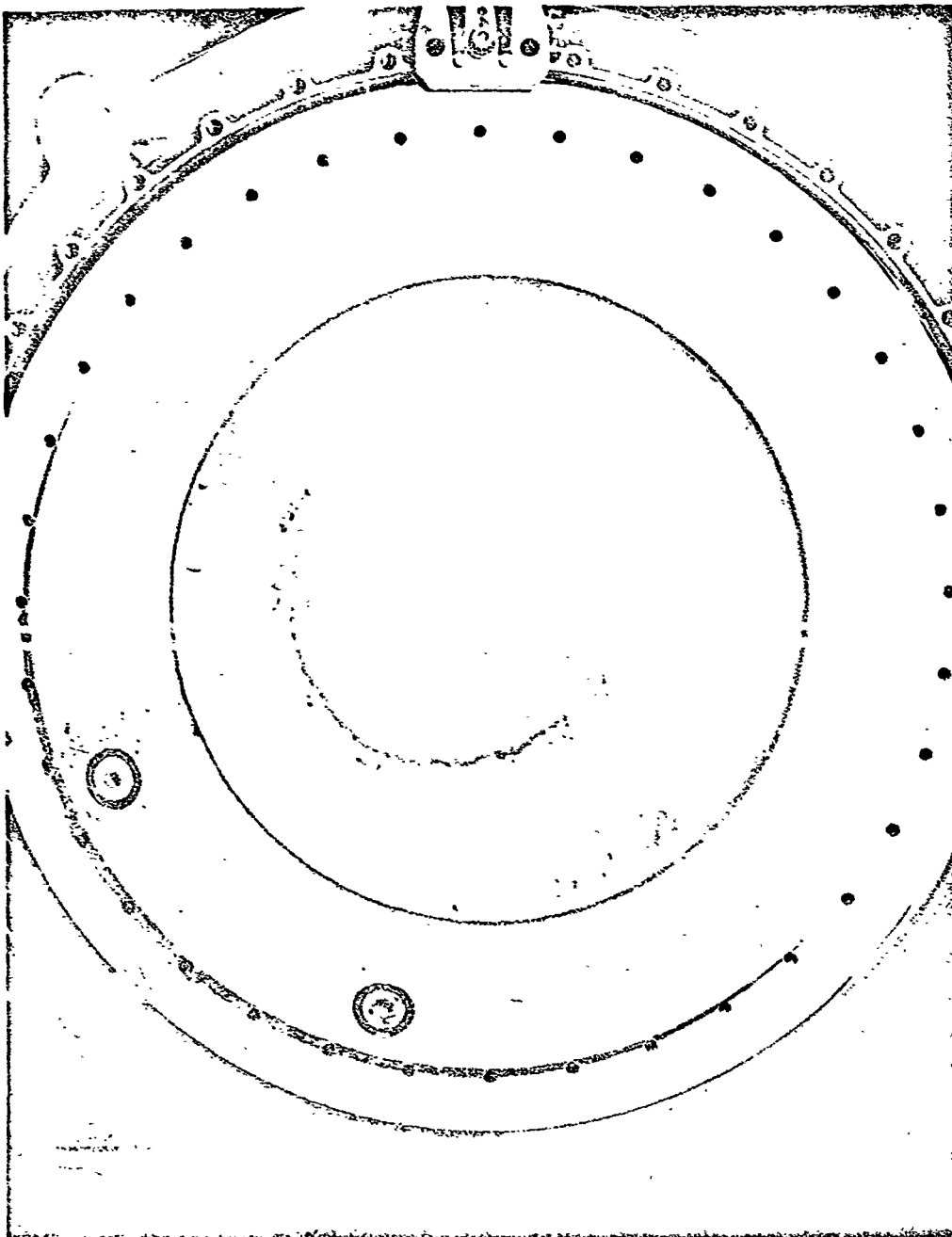


Figure 5-49. Qual B Quartz Throat Erosion, View From Forward End

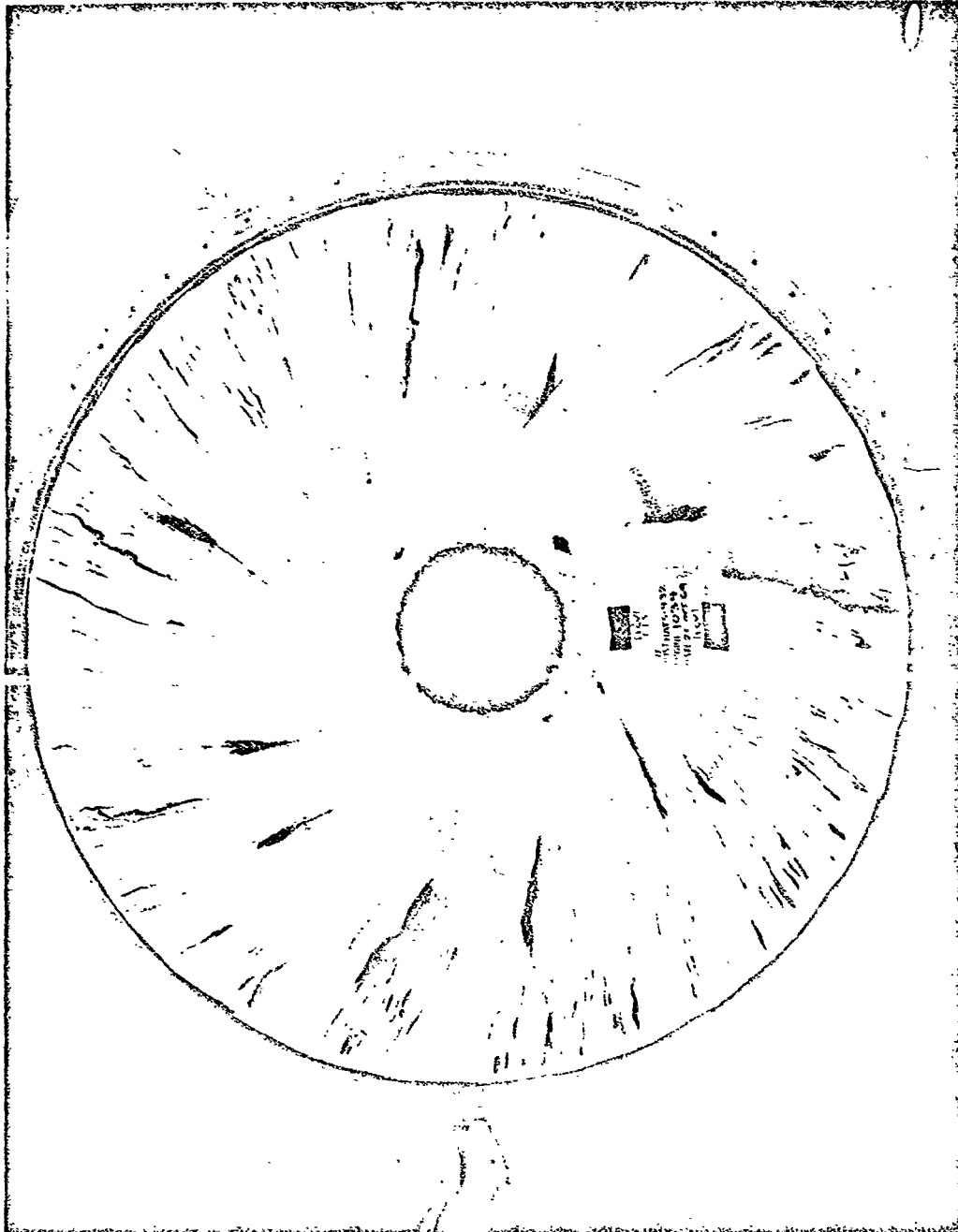
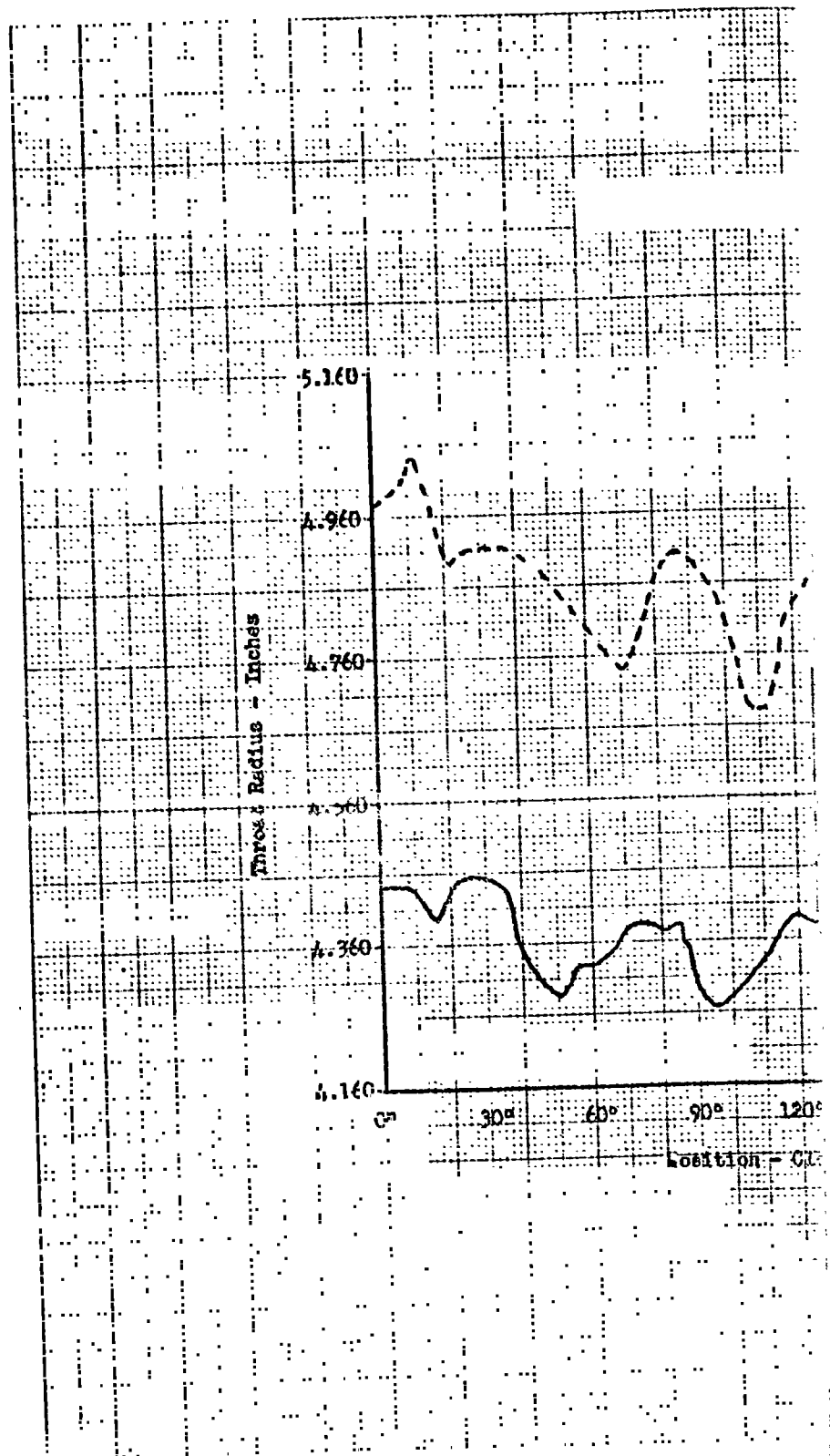
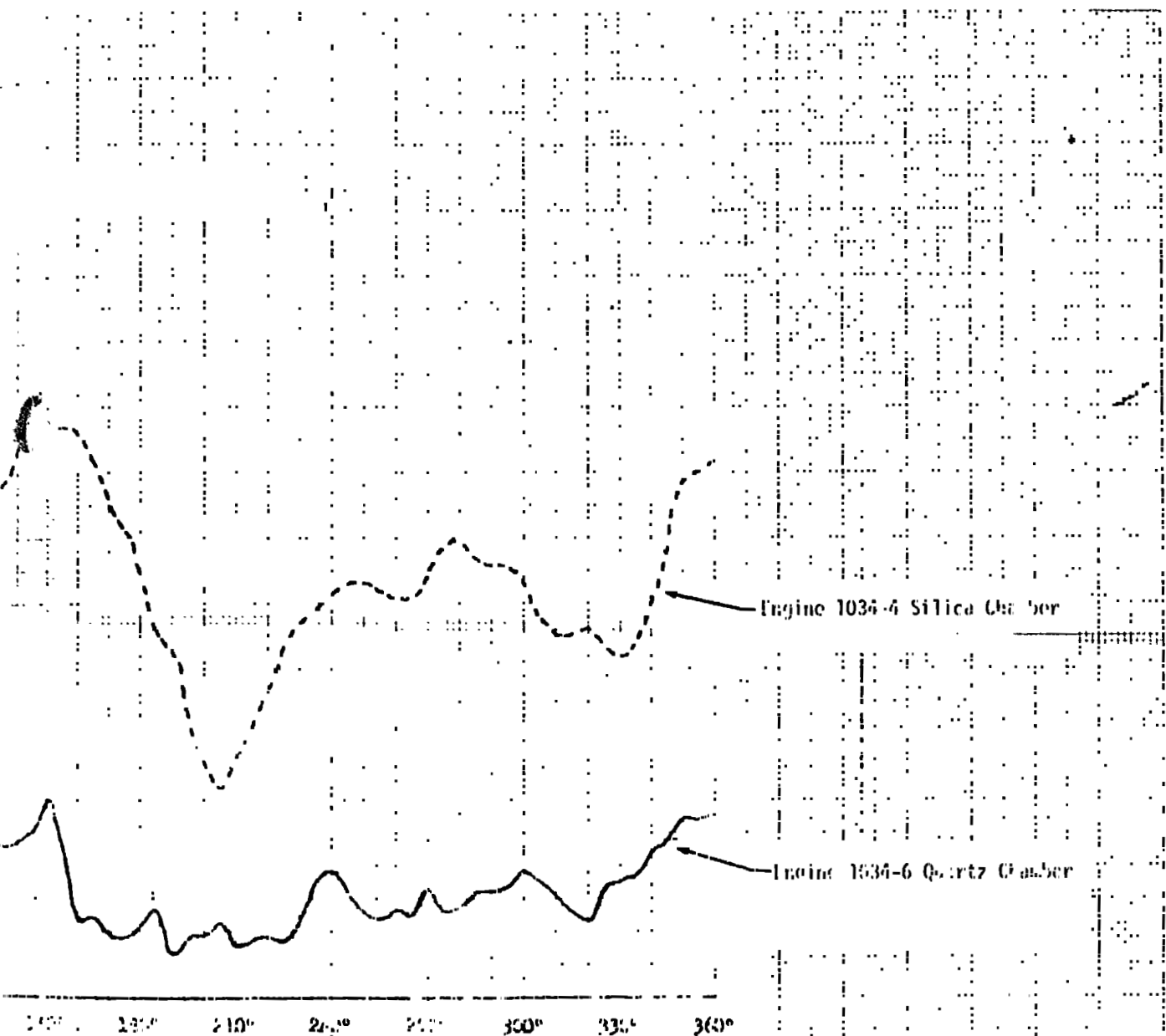


Figure 5-50. Qual B Quartz Chamber Erosion, View from Aft End



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View From Exhaust - 180°

Eng 1034

Test No.	Time	Date	Duration	Location	Station
ENG-397 - 362	1.00	6/14/49- 6/22/49	1177 hrs	3.53	39.59
ENG-426 - 432	1.00	10/12/49- 10/20/49	1145 hrs	4.53	46.59

FOLDOUT FRAME 2

FIGURE 5-51. Silica Versus Quartz Erosion Profiles

When sectioned along the planes of minimum and maximum erosion, Figures 5-52 through 5-54, chamber integrity is evidenced by the uniform char and good condition of the joint. Again, fabrication imperfections did not appear to affect performance adversely.

The char and erosion patterns are presented in Figure 5-55.

The backwall temperature history at the usually hottest chamber station is compared with that of the Qual B chamber in Figure 5-56. This plot reflects the effects of the interrupted firing. It is apparent that if the interruption had not occurred, and the final phase of the duty cycle had started at 250-300°F, as is usual following the one hour soak-back, that shutdown temperatures would have been equivalent.

Too, had not the firing been interrupted, it is estimated that total throat erosion would have been about 12 percent, instead of the 8 percent measured.

Following the duty cycle, the HEA was removed from the chamber assembly and an FTP streak test, VA1-747, was performed. The T_{120} calculated from 15 and 58 second data points was 85.1 seconds at an average mixture ratio of 1.584. When corrected to a 1.600 mixture ratio, the T_{120} was 78.7 seconds, which agrees quite well with the corrected pre-duty-cycle value of 81.5 seconds (VA2-698).

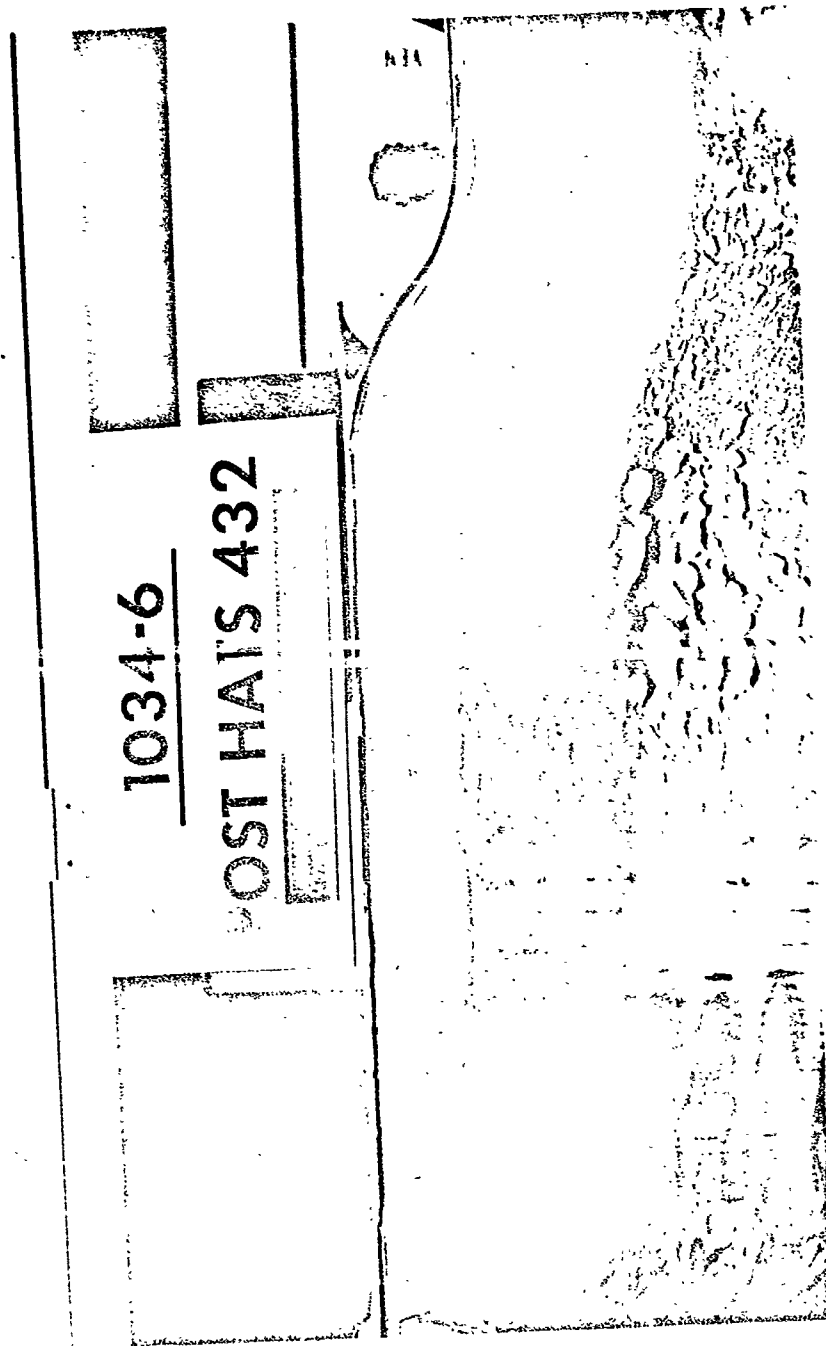


Figure 5-52. Minimum Erosion Plane

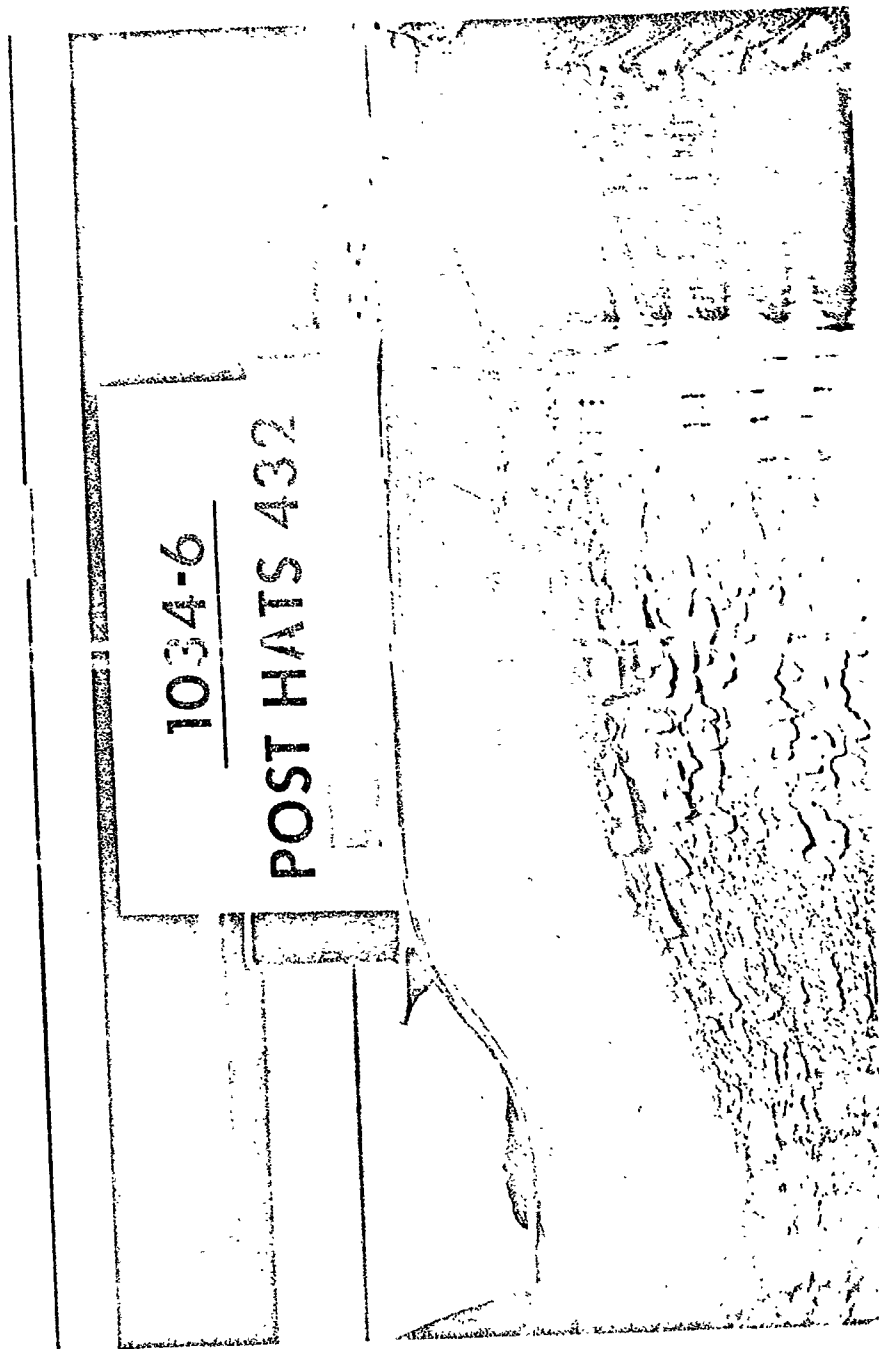


Figure 5-53. Maximum Erosion Plane

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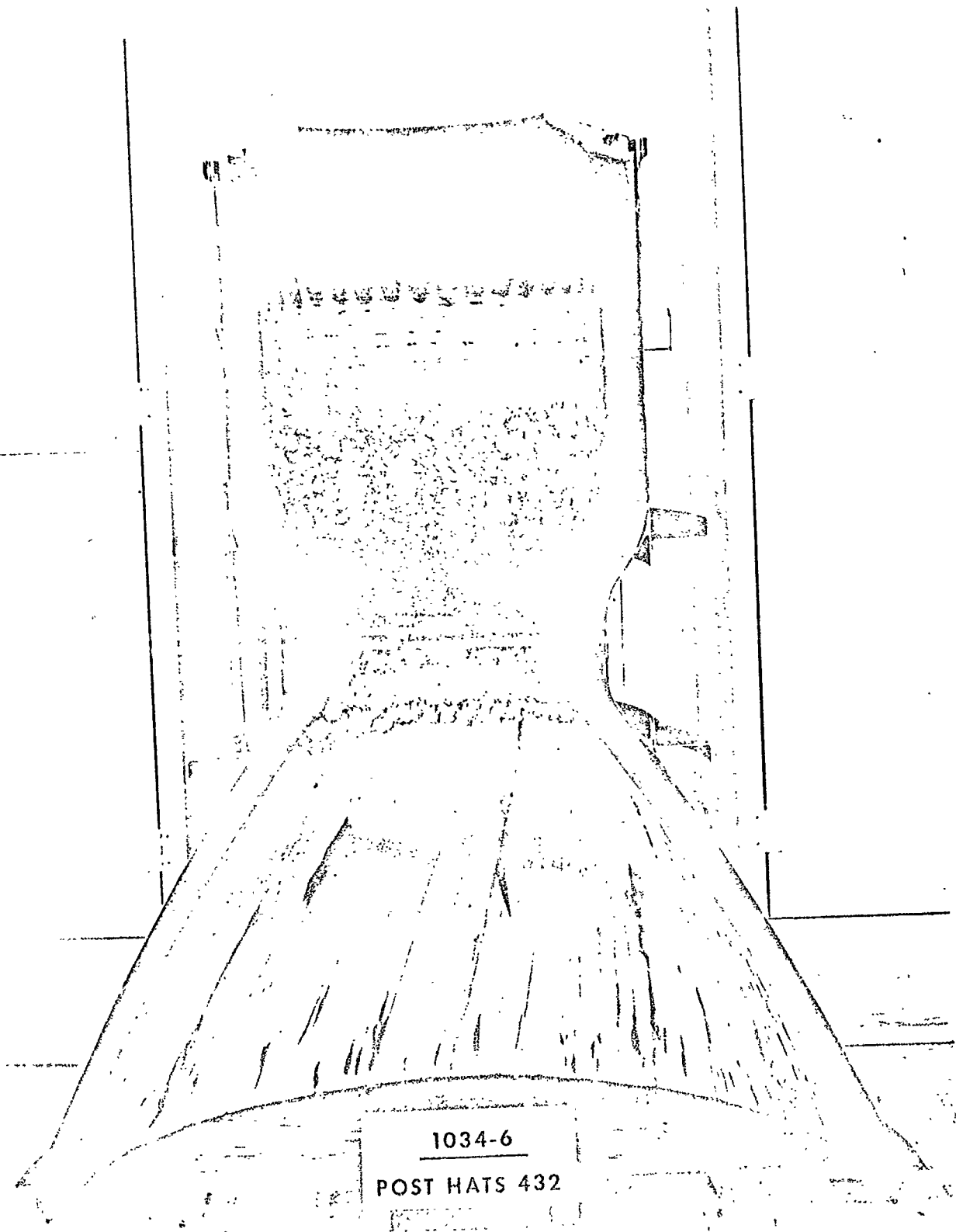


Figure 5-54. Sectioned Chamber

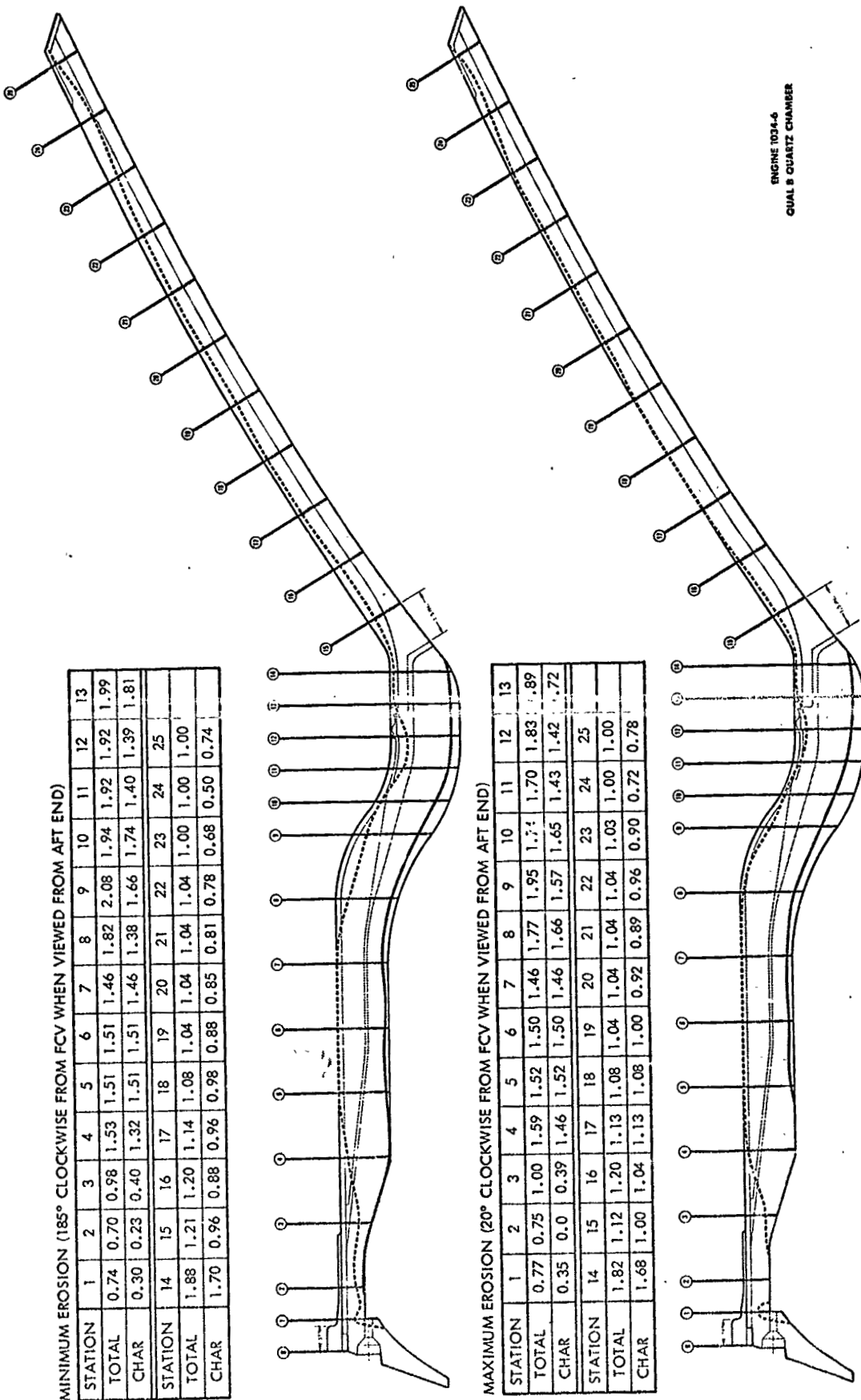


Figure 5-55. Qual B Quartz Chamber Char Pattern

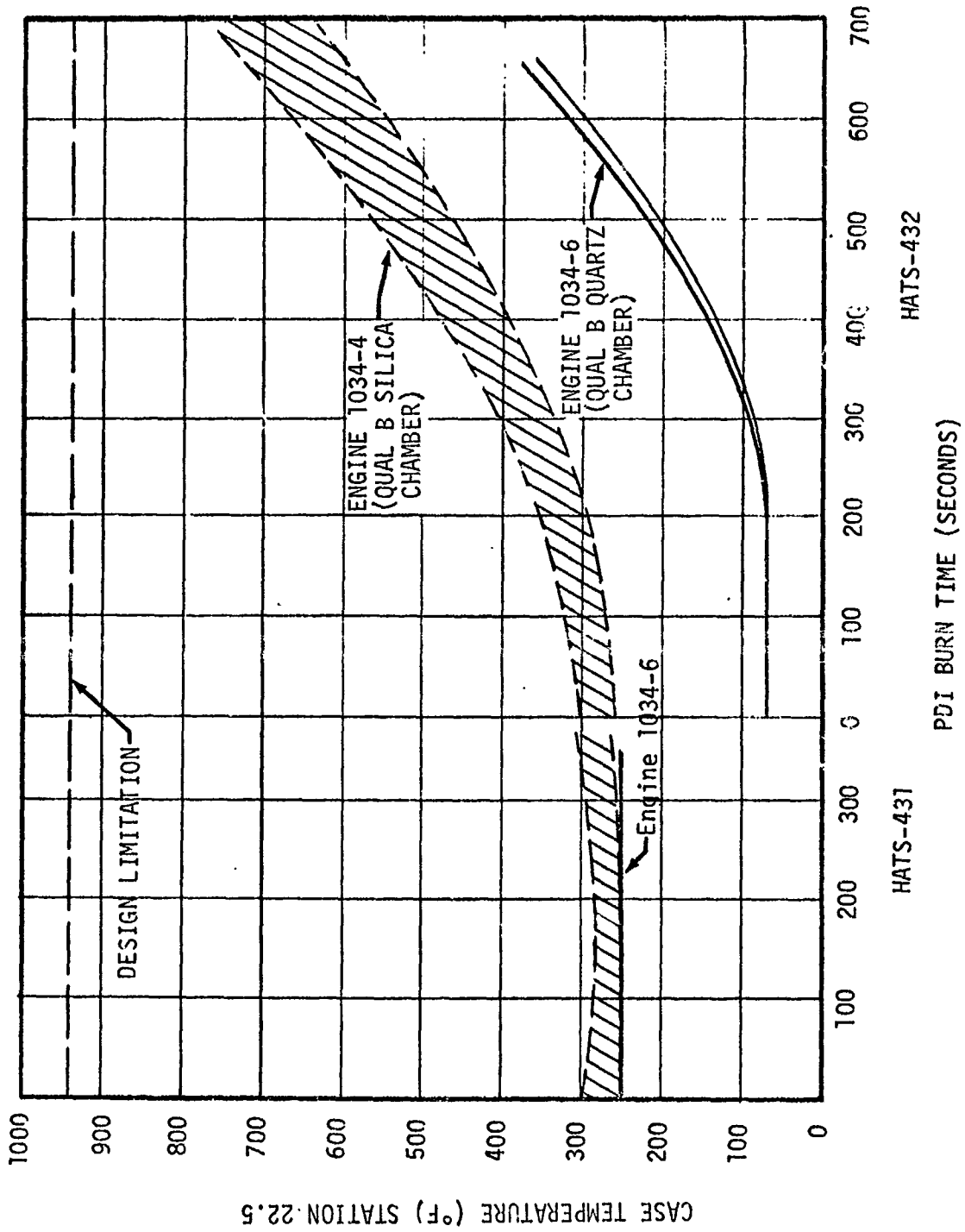


FIGURE 5-56. LONG TANK PROGRAM CASE TEMPERATURE MEASUREMENTS

6.0 CONCLUSIONS AND RECOMMENDATIONS

Conclusions pertinent to the three phases of the program are contained in the individual sections of this report. Those of significance are repeated among the general observations of this section.

The principal conclusion, the one that resulted in direct benefit to the Apollo program, was the superior erosion resistance of quartz-fabric/phenolic over its silica counterpart in the LMDE chamber environment. In all cases tested, including the Qual B configuration in an extended G mission duty cycle, the quartz erosion was about half that of silica fired under similar test conditions.

One chamber configuration evaluated in the program, Contour A, resulted in a significant reduction in heat flux and throat erosion at the full-throttle position. If the need should arise for a non-throttling duty cycle, or one conducted primarily at full thrust, a Contour A chamber would certainly offer a distinct advantage in greatly reduced erosion and the concomitant overall performance.

Despite the significant reduction in erosion experienced when silica fabric was replaced with quartz fabric in the LMDE chamber and throat, further potential ablative material advances for the nitrogen tetroxide and hydrazine type of fuel appear to be limited. One potential that should be investigated further is that of in situ formation of silicon carbide by endothermic reactions in the ablative structure. The phenomenon was explored initially in this program, but additional evaluation of the variations in the silica/carbon ratio and methods of increasing the proximity (and reactivity) of the two could prove fruitful.

Although bordering on the hard-throat category, the metal-infiltrated precharred graphitic laminates warrant further investigation. Although these have apparently overcome the poor thermal shock resistance of the refractories while exhibiting good erosion resistance, their high thermal conductivity requires a sophisticated design approach to accommodate thermal gradients and maintain the pressure vessel structure at reasonable temperatures.

The interaction of a complex two-zone combustion flow field, in which the primary and secondary oxidizer streams react with the fuel sheet and the barrier coolant, points up the fact that the zone nearest the wall at the throat is significantly detrimental to the integrity of ablative material during the present LMDE duty cycle firing sequence.

Wall environmental control can reduce erosion. In this program, it was accomplished successfully by reduction of the turbulator diameter. The turbulator improves the flow uniformity in the LMDE oxidizer jets and decreases the amount of oxidizer penetration to the chamber wall. Unfortunately, the reduced erosion was accompanied by a reduction in specific impulse.

Other methods of wall environmental control that warrant investigation include:

- Changing the oxidizer-to-fuel pressure drop ratio by adjustments in the fuel-metering ramp on the injector control sleeve.
- Varying the fuel barrier coolant flow with throttle setting.

Certain types of test methods developed or refined during this program can probably be applied advantageously in future similar programs. They include:

- An inexpensive torch test developed for screening materials that permits erosion rating of candidates relative to a standard (Section 2.3). Propellant exhaust gas composition and heat flux can be simulated. The technique permits judicious selection of materials, usually expensive, for testing in the actual engine reactive environment.
- A method of fabricating and testing a multi-material throat or chamber section (Section 5.3). The composite, with individual sections bonded along radial planes, permits comparison of erosion and thermal characteristics at any axial station relative to a standard material exposed to the same reactive test environment.

- A rapid and inexpensive data reduction technique by which heat sink data were processed and analyzed on the test-site computer (Section 3.5.3). Constant gas recovery temperature and convective heat transfer coefficient were printed out from a least square curve fit of raw wall temperature and time data. The technique is now in general use in other TRW engine test programs, replacing a more costly and time consuming numerical solution of the temperature diffusion equation on an IBM 7094 computer.

Finally, the results of the program demonstrate the desirability of low-level technology programs, even on production components. Such programs serve a two-fold purpose of maintaining: a) the technical proficiency of a developmental staff, and b) the ability to incorporate developed improvements rapidly into the production item should more stringent requirements arise.

TABLE 7-1

11602-6039-R0-00

CIC ACCUMULATIVE TEST SUMMARY

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
CALIBRATION TESTS					
VAL-655	28 Oct 68	32	32 HEA 023	Water-cooled Qual "B"	Checkout Test - Low M.R.
VAL-656	29 Oct 68	42	74	Water-cooled Qual "B"	Checkout Test - On M.R.
VAL-657	30 Oct 68	62	136	Streak Qual "B"	T_{120} = 69 seconds 1.025 turb.
VAL-658	1 Nov 68	62	198	Streak Qual "B"	T_{120} = 89 seconds Low M.R. - 1.000 turb.
VAL-659	5 Nov 68	62	260	Streak Qual "B"	T_{120} = 79 seconds 1.000 turb. - Repeat of VAL-658 - On M.R.
VAL-660	8 Nov 68	58	318	Water-cooled Qual "B"	Acceptance Test 65 psia GN_2 -Saturated Propellants - On M.R.
VAL-661	8 Nov 68	58	376	Water-cooled Qual "B"	Acceptance Test 65 psia GN_2 -Saturated Propellants - On M.R.
VAL-662	12 Nov 68	58	434	Water-cooled Qual "B"	Acceptance Test 240 psia He -Saturated Propellants
VAL-663	14 Nov 68	62	496	Streak Qual "B"	T_{120} = 74 seconds 240 psia He -Saturated Propellants
VAL-664	19 Nov 68	102	598	Streak Qual "B"	25% F - Low M.R.
VAL-665	20 Nov 68	106	704	Streak Qual "B"	25% F - T_{120} = 152 seconds

TABLE 7-1

11602-6039-RO-00

CIC ACCUMULATIVE TEST SUMMARY (Continued)

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
EROSION BASELINE TESTS					
VA1-666	25 Nov 68	26	730	2:1 Ablative	Checkout Test High M.R.
VA1-667	4 Dec 68	51	781	2:1 Ablative	Checkout Test Low M.R.
VA1-668	5 Dec 68	22	803	2:1 Ablative	Aborted due to loose "B" nut on oxidizer instrumen- tation line
VA1-669	5 Dec 68	51	854	2:1 Ablative	Checkout Test On M.R.
VA1-670	5 Dec 68	42	896	2:1 Ablative	Duty Cycle (Phase I)
VA1-671	5 Dec 68	822	1718	2:1 Ablative	Duty Cycle (Phase II)

BASELINE CHAMBER THERMAL CHARACTERIZATION
(HEAT SINK CHAMBER)

VA1-672	11 Dec 68	33	1751	Water-Cooled Qual "B"	Checkout Test
VA1-673	19 Dec 68	7	1758	Qual "B" Heat Sink	FTP Test
VA1-674	19 Dec 68	8	1766	Qual "B" Heat Sink	FTP Test
VA1-675	27 Dec 68	17	1783	Qual "B" Heat Sink	25% F Test
VA1-676	27 Dec 68	22	1805	Qual "B" Heat Sink	50% F Test
VA1-677	27 Dec 68	10	1815	Qual "B" Heat Sink	65% F Test
VA1-678	27 Dec 68	13	1828	Qual "B" Heat Sink	65% F Test

TABLE 7-1

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CIC ACCUMULATIVE TEST SUMMARY (Continued)

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
VA1-679	27 Dec 68	29	1857	Qual "B" Heat Sink	25% F Test
VA1-680	27 Dec 68	31	1868	Qual "B" Heat Sink	50% F Test
VA1-688	17 Jan 69	42	1930	Qual "B" Heat Sink	25% F Test - chamber rotated relative to the HEA
VA1-689	17 Jan 69	62	1992	Qual "B" Heat Sink	10% F Test
VA1-690	17 Jan 69	12	2004	Qual "B" Heat Sink	65% F Test
VA1-691	17 Jan 69	25	2029	Qual "B" Heat Sink	50% F Test
VA1-692	28 Jan 69	52	2081	Qual "B" Heat Sink	25% F Test
VA1-693	28 Jan 69	9	2090	Qual "B" Heat Sink	FTP Test
VA1-694	28 Jan 69	42	2132	Qual "B" Heat Sink	25% F Test No barrier coolant
MODIFIED CHAMBER CONTOUR					
VA1-695	18 Feb 69	10	2142	Contour "A" Fiberglass Liner	FTP - High M.R.
VA1-696	18 Feb 69	16	2158	Contour "A" Fiberglass Liner	FTP - On M.R.
VA1-697	18 Feb 69	60	2218	Contour "A" Fiberglass Liner	25% F Test
VA1-698	19 Feb 69	51	2269	Contour "A" Fiberglass Liner	25% F Test-Liner Glass Flow Occurred

TABLE 7-1

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CIC ACCUMULATIVE TEST SUMMARY (Continued)

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
VA1-699	11 Mar 69	14	2283	Contour "A" Steel Liner	FTP Test
VA1-700	11 Mar 69	52	2335	Contour "A" Steel Liner	25% F Test
VA1-701	11 Mar 69	27	2362	Contour "A" Steel Liner	50% F Test
VA1-702	11 Mar 69	17	2379	Contour "A" Steel Liner	65% F Test
VA1-704	18 Mar 69	17	2396	Contour "B" MX 2600 Liner	FTP Test
VA1-705	18 Mar 69	52	2448	Contour "B" MX 2600 Liner	25% F Test
VA1-706	18 Mar 69	28	2476	Contour "B" MX 2600 Liner	50% F Test
VA1-707	18 Mar 69	18	2494	Contour "B" MX 2600 Liner	65% F Test
VA1-708	18 Mar 69	15	2509	Contour "B" MX 2600 Liner	FTP Test - Repeat of VA1-704 with slightly higher M.R.
VA1-709	19 Mar 69	9	2518	Contour "C" MX 2600 Liner	FTP Test
VA1-710	20 Mar 69	11	2529	Contour "C" MX 2600 Liner	FTP Test
VA1-711	20 Mar 69	52	2581	Contour "C" MX 2600 Liner	25% F Test
VA1-712	20 Mar 69	28	2609	Contour "C" MX 2600 Liner	50% F Test
VA1-713	20 Mar 69	12	2621	Contour "C" MX 2600 Liner	65% F Test

TABLE 7-1

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CIC ACCUMULATIVE TEST SUMMARY (Continued)

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
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IMPROVED BARRIER COOLANT FLOW RATE

VA1-714	21 Mar 69	9	2630	Contour "C" MX 2600 Liner	FTP Test - Increased barrier coolant flow - low M.R.
VA1-715	21 Mar 69	8	2638	Contour "C" MX 2600 Liner	FTP Test - Increased barrier coolant flow
VA1-716	21 Mar 69	7	2645	Contour "C" MX 2600 Liner	FTP Test - Increased B.C. flow - 0.960 dia. turb.

MODIFIED CHAMBER CONTOUR TESTS
(Continued)

VA1-717	27 Mar 69	62	2707	Contour "C" Fiberglass Streak - MX 2600 Liner	FTP Test - T ₁₂₀ = 80 seconds Low M.R. Nominal B.C. Flow - 1.000 dia. turb.
VA1-718	8 Apr 69	7	2714	Contour "A" Fiberglass Streak - MX 2600 Liner	FTP Test - Shutdown at 7 seconds due to high M.R. Nominal B.C. Flow - 1.000 dia. turb.
VA1-719	8 Apr 69	63	2777	Contour "A" Fiberglass Streak - MX 2600 Liner	FTP Test - Tank pressures biased from previous test to arrive at proper M.R.

BASELINE CHAMBER THERMAL CHARACTERIZATION
(WATER COOLED HEAT TRANSFER CHAMBER)

VA2-639	28 Apr 69	32	2809	Water-cooled Heat Transfer Chamber	Throttling checkout test - Off M.R.
VA2-640	28 Apr 69	178	2987	Water-cooled Heat Transfer Chamber	7 point throttling test FTP, 65, 60, 50, 40, 25, and 10% thrust
VA2-641	28 Apr 69	94	3081 HEA 023	Water-cooled Heat Transfer Chamber	5 point throttling test 10, 25, 40, 50, and 60% thrust - shutdown pre- maturely due to hot gas leak.

TABLE 7-1

11602-6039-RC -00

CIC ACCUMULATIVE TEST SUMMARY (Continued)

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
IMPROVED INJECTOR FLOW DISTRIBUTION TESTS					
VA2-642	12 May 69	8	8 XX2 Injector	Contour "A" Fiberglass Streak - MX 2600 Liner	FTP Test - Shutdown at 8 seconds due to high M.R. (1617)
VA2-643	12 May 69	60	68	Contour "A" Fiberglass Streak - MX 2600 Liner	FTP Test - T ₁₂₀ = 98.5 seconds Nominal B.C. Flow 1.025 dia. turb.
VA2-644	21 May 69	11	79	Contour "A" Heat Sink MX 2600 Liner	FTP Test 1.025 dia. turb.
VA2-645	23 May 69	18	97	Contour "A" Heat Sink MX 2600 Liner	65% Thrust High M.R. (1.643) 1.025 dia. turb.
VA2-646	23 May 69		111	Contour "A" Heat Sink MX 2600 Liner	65% Thrust Low M.R. (1.577) 1.025 dia. turb.
VA2-647	23 May 69	13	124	Contour "A" Heat Sink MX 2600 Liner	65% Thrust 1.025 dia. turb.
VA2-648	23 May 69	11	135	Contour "A" Heat Sink MX 2600 Liner	FTP Test 1.000 dia. turb.
VA2-649	23 May 69	13	148	Contour "A" Heat Sink MX 2600 Liner	FTP Test .925 dia. turb.
VA2-650	23 May 69	17	165	Contour "A" Heat Sink MX 2600 Liner	65% Thrust Low M.R. (1.546) .925 dia. turb.
VA2-651	26 May 69	13	178	Contour "A" Heat Sink MX 2600 Liner	65% Thrust .925 dia. turb.
VA2-652	26 May 69	14	192	Contour "A" Heat Sink MX 2600 Liner	FTP Test .960 dia. turb.

TABLE 7-1

CIC ACCUMULATIVE TEST SUMMARY (Continued)

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
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ADVANCED MATERIALS EROSION EVALUATION TESTS
(COMPOSITE CHAMBER)

VA1-733	3 June 69	26	3107 HEA 023	Contour "C" Composite 2:1 Ablative	Checkout Test
VA1-734	3 June 69	39	3146	Contour "C" Composite 2:1 Ablative	Duty Cycle (Phase I)
VA1-735	3 June 69	352	3498	Contour "C" Composite 2:1 Ablative	Duty Cycle (FTP burn of Phase II)
VA1-736	5 June 69	26	3524	Contour "C" Composite 2:1 Ablative	Checkout test prior to the throttling portion of Phase II of the duty cycle.
VA1-737	5 June 69	561	4085	Contour "C" Composite 2:1 Ablative	Phase II of the composite 2:1 ablative duty cycle

IMPROVED INJECTOR FLOW DISTRIBUTION TESTS
(Continued)

VA2-653	10 June 69	10	202 XX2 Injector	Contour "C" Heat Sink MX-2600 Liner	FTP Test 0.925 in. dia. Turb. Low M.R.
VA2-654	10 June 69	9	211	Contour "C" Heat Sink MX-2600 Liner	FTP Test 0.925 in. dia. turb. MR = 1.597
VA2-655	10 June 69	9	219	Contour "C" Heat Sink MX-2600 Liner	FTP Test 1.000 in. dia. turb.
VA2-656	11 June 69	60	279	Streak Contour "C"	T ₁₂₀ = 108 seconds 0.925 in. dia. turb.

MODIFIED CHAMBER CONTOUR TESTS
(Continued)

VA1-741	16 June 69	26	4111 HEA 023	2:1 Ablative Contour "A"	Checkout Test
VA1-742	16 June 69	39	4150	2:1 Ablative Contour "A"	Duty Cycle (Phase I)
VA1-743	16 June 69	819	4969	2:1 Ablative Contour "A"	Duty Cycle (Phase II)

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CIC ACCUMULATIVE TEST SUMMARY (Continued)

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
PHASE III PROTOTYPE CHAMBER TESTS					
HEA 034					
VA2-688	8 Sept 69	62	For this series 62	Qual "B" Streak Test T_{120} = 85.7 sec	T_{120} on HEA 034 prior to MDC (HATS 387-392) was VA2-675 with a T_{120} = 86.6 seconds
VA2-689	9 Sept 69	62	124	Contour "C" Streak Test T_{120} = 77.8 sec	With 1.000 inch diameter turbulator, 4.9 throttle arm ratio
VA2-690	12 Sept 69	77	201	Qual "B" Water- cooled Checkout Test	Throttle arm ratio = 4.63; 1.000 inch diameter turbulator
VA2-691	12 Sept 69	79	280	Qual "B" Water- cooled Checkout Test	Flow data invalid because engine line recirculation valves were open during test. 0.925 inch diameter turbulator.
VA2-692	12 Sept 69	79	359	Qual "B" Water- cooled Checkout Test	0.925 inch diameter turbu- lator 4.63 throttle arm ratio
VA2-693	15 Sept 69	62	421	Contour "C" Streak Test T_{120} = 108.4 sec	0.925 inch diameter turbu- lator 4.63 throttle arm ratio
HATS-416	29 Sept 69	31	452	Contour "C" Quartz Chamber Assembly	65 psia GN_2 checkout test; 1.000 inch diameter turbulator
HATS-417	30 Sept 69	43	495	Contour "C" Quartz Chamber Assembly	240 psia helium character- ization test; 1.000 inch diameter turbulator.
HATS-418	30 Sept 69	25	520	Contour "C" Quartz Chamber Assembly	0.925 inch diameter turbu- lator; 240 psia He- saturated propellants.

TABLE 7-1

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CIC ACCUMULATIVE TEST SUMMARY

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
HATS-419	2 Oct 69	22	542	Contour "C" Quartz Chamber Assembly	Calibration Test
HATS-420	3 Oct 69	35	577	Contour "C" Quartz Chamber Assembly	Acceptance Test
HATS-421	3 Oct 69	28	605	Contour "C" Quartz Chamber Assembly	Duty Cycle (Phase I)
HATS-422	3 Oct 69	985	1590	Contour "C" Quartz Chamber Assembly	Duty Cycle (Phase II)
VA2-698	6 Oct 69	62	1652	Streak Qual "B"	$T_{120} = 79.6$ seconds High MR corrected to 82.8 seconds 1.000 dia. turb.
HATS-426	13 Oct 69	22	1674	Qual "B" Con- tour Quartz Chamber Assem- bly	Calibration Test with 1.000 dia. turb.
HATS-427	15 Oct 69	40	1714	Qual "B" Con- tour Quartz Chamber Assembly	Acceptance Test - 65 psia GN_2 Saturated Propellants
HATS-428	15 Oct 69	43	1757	Qual "B" Con- tour Quartz Chamber Assembly	Acceptance Test - 65 psia GN_2 Saturated Propellants
HATS-429	15 Oct 69	43	1800	Qual "B" Contour Quartz Chamber Assembly	Acceptance Test - 240 psia Helium Saturated Propellants

TABLE 7-1

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CIC ACCUMULATIVE TEST SUMMARY

Test No.	Date	Duration (Seconds)	Element Accumulative Duration (Seconds)	Chamber Configuration	Comments
HATS-430	16 Oct 69	28	1828	Qual "B" Con- tour Quartz Chamber Assembly	Duty Cycle (Phase I)
HATS-431	16 Oct 69	346	2174	Qual "B" Con- tour Quartz Chamber Assembly	Duty Cycle (Phase II) Aborted When Facility S.O.V. Cable Shorted Out
HATS-432	20 Oct 69	688	2862	Qual "B" Con- tour Quartz Chamber Assembly	Continuation of Duty Cycle
VAL-747	24 Oct 69	62	2924	Streak Qual "B"	T ₁₂₀ = 85.1 second Low MR - Corrected to 82.7 seconds

8.0 REFERENCES

<u>NUMBER</u>	<u>CIC REPORT NUMBER</u>	<u>TITLE</u>	<u>DATE</u>
1	11602-6001-T0-00	Compatibility Improvement Study Planning Report	26 July 1968
2	11602-6002-R0-00	Quality Program Plan	2 August 1968
3	11602-6003-R0-00	Monthly Progress Report	10 August 1968
4	11602-6005-R0-00	Monthly Progress Report Supplement	16 August 1968
5	11602-6006-R0-00	Monthly Progress Report	10 September 1968
6	11602-6007-T0-00	LMDE Qual B Head End Assembly Reactive Calibration Test Plan	17 September 1968
7	11602-6008-R0-00	Design Review Data Package	20 September 1968
8	11602-6010-R0-00	Third Monthly Progress Report	8 October 1968
9	11602-6009-T0-00	Water Cooled Chamber Design Review Data Package	4 October 1968
10	11602-6011-T0-00	LMDE Qual B Head End Assembly Erosion Baseline Test Plan (2:1 Chamber)	10 October 1968
11	11602-6012-R0-00	Fourth Monthly Progress Report	12 November 1968
12	11602-6013-T0-00	Baseline Chamber Thermal Characterization Test Plan	13 November 1968
13	11602-6015-R0-00	Fifth Monthly Progress Report	12 December 1968
14	11602-6016-T0-00	Modified Chamber Contour Thermal Characterization Test Plan	18 December 1968
15	11602-6017-T0-00	Reactive Calibration Test Report	7 January 1969
16	11602-6018-R0-00	Sixth Monthly Progress Report	13 January 1969
17	11602-6019-T0-00	Water Cooled Chamber Heat Transfer Test Plan	20 January 1969
18	11602-6020-T0-00	Erosion Baseline Test Report (2:1 Chamber)	28 January 1969

REFERENCES (CONTINUED)

<u>NUMBER</u>	<u>CIC REPORT NUMBER</u>	<u>TITLE</u>	<u>DATE</u>
19	11602-6020-T0-01	Erosion Baseline Test Report (2:1 Chamber) Revised	31 March 1969
20	11602-6021-R0-00	Seventh Monthly Progress Report	12 February 1969
21	11602-6022-R0-00	Materials Survey	3 March 1969
22	11602-6024-T0-00	Eighth Monthly Progress Report	12 March 1969
23	11602-6023-T0-00	Improved Barrier Coolant Flow Rate Test Plan	12 March 1969
24	11602-6025-T0-00	Advanced Materials Erosion Evaluation Test Plan	2 April 1969
25	11602-6026-R0-00	Torch Test Materials Evaluation	11 April 1969
26	11602-6027-R0-00	Ninth Monthly Progress Report	18 April 1969
27	11602-6028-T0-00	Improved Injector Flow Distribution Test Plan	22 April 1969
28	11602-6029-R0-00	Tenth Monthly Progress Report	12 May 1969
29	11602-6030-T0-00	Modified Chamber Contour Test Plan (2:1 Ablative)	28 May 1969
30	11602-6031-R0-00	Eleventh Monthly Progress Report	12 June 1969
31	11602-6031-T0-00	Modified Chamber Contour Test Plan (2:1 Ablative Contour "C")	12 June 1969
32	11602-6033-R0-00	Twelfth Monthly Progress Report	15 July 1969
33	11602-6034-R0-00	Thirteenth Monthly Progress Report	30 August 1969
34	11602-6035-T0-00	Phase III Prototype Chamber Test Plan	24 September 1969
35	11602-6036-R0-00	Fourteenth Monthly Progress Report	30 September 1969
36	11602-6037-R0-00	Fifteenth Monthly Progress Report	15 October 1969
37	01827-6277-T0-00	Final Test Report, Long Tank Program, CIC "C" Quartz Chamber	15 December 1969