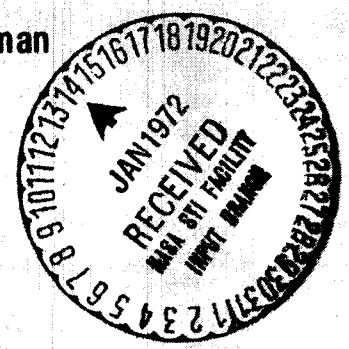# APPLICATION OF ABLATION TO A
HIGH CHAMBER PRESSURE ROCKET ENGINE

by

A. W. Brecheisen and C. M. Ehresman

PURDUE UNIVERSITY

JET PROPULSION CENTER

School of Mechanical Engineering

Lafayette, Indiana 47907

prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

1. Report No. NASA CR-120798	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle APPLICATION OF ABLATION TO A HIGH CHAMBER PRESSURE ROCKET ENGINE		5. Report Date August, 1971
		6. Performing Organization Code
7. Author(s) A. W. Brecheisen and C. M. Ehresman		8. Performing Organization Report No.
		10. Work Unit No.
9. Performing Organization Name and Address Purdue University Jet Propulsion Center School of Mechanical Engineering Lafayette, Indiana 47907		11. Contract or Grant No. NGR 15-005-058
		13. Type of Report and Period Covered Contractor Report
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		14. Sponsoring Agency Code

15. Supplementary Notes

Technical Officer, Richard J. Priem, Chemical Rocket Division, NASA Lewis Research Center,

Cleveland, Ohio

16. Abstract

Ablation cooling was successfully used in a rocket engine at a nominal chamber pressure of 4000 psia (272.2 atm). Both the chamber liner and the nozzle were constructed of a silica phenolic ablative material. Linear ablation rates at the nozzle throat for thirty-six firings were computed from run data. The overall average rate obtained was 0.0861 in/sec (0.219 cm/sec). The linear ablation rate obtained for the chamber liner was 0.071 in/sec (0.180 cm/sec) at the nozzle entrance end.

17. Key Words (Suggested by Author(s)) Ablation High Chamber Pressure Rocket Engine	18. Distribution Statement Unclassified - unlimited		
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 20	22. Price* $3.00

* For sale by the National Technical Information Service, Springfield, Virginia 22151

ABSTRACT

Ablation cooling was successfully used in a rocket engine at a nominal chamber pressure of 4000 psia (272.2 atm). Both the chamber liner and the nozzle were constructed of a silica phenolic ablative material. Linear ablation rates at the nozzle throat for thirty-six firings were computed from run data. The overall average rate obtained was 0.0861 in/sec (0.219 cm/sec). The linear ablation rate obtained for the chamber liner was 0.071 in/sec (0.180 cm/sec) at the nozzle entrance end.

I. SUMMARY

Ablation cooling was successfully used to cool a rocket engine at high chamber pressure (norminal 4000 psia, 272.2 atm). The purpose of the research was to investigate the performance and the combustion stability characteristics of the engine. Therefore, a comprehensive analysis of ablation was not accomplished. Instead, the present report is concerned with a qualitative presentation of the results of this application of a specific ablative material.

The material used was a silica phenolic ablative. Both the combustion chamber liners and the nozzles were constructed of this material. As part of the performance and stability studies it was necessary to measure prefire and postfire nozzle throat diameters. Since run durations were known as well for each firing, it was possible to calculate corresponding linear ablation rates from these data.

Data reported herein include run duration, mixture ratio, chamber pressure, pulse-no-pulse, injector tube diameters, number of firings on a nozzle, runs numbers corresponding to the firings on a nozzle, and the nozzle throat linear ablation rate. These data were analyzed using several different methods.

Examinations of cross-sections of the nozzles and chamber liners, after use during firings of the engine, revealed a significant difference in their respective char layers. The char layers extended relatively deeply into the walls of the chamber liners and the inside walls were coated with silica melt. In contrast, the char layers in the nozzle cross-sections were very thin and no silica melt could be seen. This difference between the char layers was attributed to the difference in silica fiber orientation for the two engine parts. The fibers were oriented parallel to the engine axis in the chamber liners whereas the fiber orientation in the nozzles was perpendicular to the axis line of the engine.

The overall average linear ablation rate obtained at the nozzle throat was 0.0861 in/sec (0.219 cm/sec). One measurement was performed on a chamber liner at the nozzle entrance end which yielded an ablation rate of 0.071 in/sec (0.180 cm/sec).

II. INTRODUCTION

The purpose of this report is to record the results of an application of ablative material in a high chamber pressure (4000 psia, 272.2 atm) rocket engine. The primary objective of the experimental program was not to evaluate the material but rather to study the performance and combustion stability of a high pressure engine. It was felt, however, that the performance of the ablatives used under these conditions might be of interest to others working in the high pressure area. The investigation of the consistency of results, described herein, was undertaken in an effort to establish a correlation between ablation rates and heat transfer.

Ablation cooling has been successfully applied to many reentry vehicle and rocket engine heat transfer problems. Ablation cooling involves the use of a protective material to cover the exterior part of a structure which is subjected to a high heat flux. The protective material thermally degrades and the products of this degradation are carried away by the external fluid flow. This type of cooling is necessarily limited in the amount of total heat flux protection it gives due to the loss of material from the surface. The thermal protection is gone once the ablative material has been consumed. The total heat flux protection offered by a given system is affected by the heat transfer mechanism, by the thickness of the ablative material, and by the properties of the material itself. There are many different materials that qualify as candidates for ablative cooling. The particular application dictates what type of material is actually used.

In the present report, attention is focused upon a particular class of ablative materials known as the charring ablator. Charring ablators generally consist of a basic structural material, such as nylon or fiberglass cloth, impregnated with some type of bonding material (e.g. phenolic resin).

III. EXPERIMENTAL METHOD

Nozzle throat ablation rates were measured in an ablatively cooled high pressure rocket engine. The ablative material used was Western Backing 2230 which is a composite consisting of silica cloth impregnated with phenol aldehyde resin. The material contains approximately thirty per cent resin by weight. The nozzle material was produced by flat stacking of layers of silica cloth upon one another, with the weave on each new layer rotated 90° to the weave of the previous layer, impregnating the cloth with resin, and then baking the composite to cure the resin. The nozzles were then machined from this material such that the plane of the silica cloth was perpendicular to the nozzle axis. The ablative chamber liners were produced in an analogous manner except that the silica cloth was wrapped around a cylindrical mandrel before impregnation and curing. The resulting liners had the plane of the silica cloth parallel to the axis line of the engine.

2

A. Apparatus

The rocket engine used developed approximately 4000 pounds (1815 kg) of thrust with a design chamber pressure of 4000 psia (272.2 atm). Figure 1 is a cross-sectional sketch of the engine. The nozzle and chamber liners were constructed of the ablative material. The engine was of the liquid bipropellant type using nitrogen tetroxide and Aerozine-50 as propellants. The injector was a 7 quadlet tube injector with like-on-like flow impingement within the individual quadlets (Figure 2). Data recording equipment utilized included eleven strip chart recorders, a twenty channel oscillograph, a digital data recording system, and an analog data recording system.

B. Test Procedures

Fifty firings of the engine were performed as part of a performance and combustion stability study. Of these fifty firings, thirty-six were successfully completed (i.e. complete run performance data were obtained). The data, presented later in this report, were taken from these thirty-six runs. The reported data include linear ablation rate, run duration, mixture ratio, chamber pressure, injector configuration, pulse-no-pulse, number of runs on each nozzle, and the engine firing numbers in which a given nozzle was used. The ensuing discussion will explain how these data were obtained.

The throat ablation rate data resulted from nozzle throat diameter measurements made both before and after each run, in order to analyze the performance data. The throat diameter was measured with a micrometer at four circumferential locations, each location being spaced at 45° to the previous measurement location. Throat ablation occurred unequally around the circumference of the throat, as shown in Figure 3 (produced by directing light through a nozzle and then through a ground glass plate, tracing the outline of the throat shown on the glass plate, and enlarging the tracing using dividers). The throat diameter used in computing ablation rate was the average of the micrometer measurements taken at the throat. The linear throat ablation rate was calculated using:

$$\text{ablation rate} = \frac{\text{postfire throat dia. - prefire throat dia.}}{2 \ (\text{run duration})} \qquad (1)$$

Run duration was determined from the oscillograph trace of chamber pressure for each run and was defined as the length of time between attainment of one-half of the steady state run chamber pressure during startup and the drop to one-half of the steady state pressure at shutdown (Figure 4).

Propellant flow rates were measured with turbine meters whose outputs were recorded on the oscillograph.

Chamber pressure was measured with a capacitance transducer and with

3

a strain gage pressure transducer. The transducer outputs were recorded on the oscillograph, the digital recording system, and the analog tape recording system. A typical oscillograph trace of the photocon chamber pressure measurement for a pulsed run is shown in Figure 4.

Other data reported were dependent upon the engine configuration for a given run and the purpose of the run. These included pulse-no-pulse, nozzle run number, and injector configuration. The engine was pulsed during some firings using a pulse gun charged with 56.4 grains of RDX waffers, as part of the combustion stability study. The resulting pulse-no-pulse data simply indicates whether the engine was or was not pulsed during a given run.

Nozzles were utilized for up to four firings each. Nozzle usage was limited by throat growth due to ablation. The engine run numbers for which a particular nozzle was used as well as the number of runs on a nozzle are reported.

Injectors used consisted of seven quadlets with like-on-like impingement (Figure 2). Two different sizes of fuel and oxidizer tubes were used as part of the performance and combustion instability programs.

C. Test Results

The data are presented in Table 1. An overall average linear ablation rate of 0.0861 in/sec (0.219 cm/sec) was obtained at the nozzle throat. The linear ablation rate was measured for one chamber liner and was found to be 0.071 in/sec (0.180 cm/sec) at the nozzle end of the liner. Pictures of cross-sections of a chamber liner and several nozzles are shown in Figure 5. Examination of the liner cross-section, 5a, reveals a discoloration zone along the inside perimeter which extends through approximately 20% of the thickness of the remaining liner wall. Partial pyrolysis of the ablative wall has occured in this region. Also shown in 5a is a thin light colored layer, at the inside perimeter of the cross-section, which is silica melt. The other three photographs, 5b, 5c, 5d, show various cross-sections of the ablated nozzle material. In contrast to the chamber liner, there is relatively little char zone and no silica melt layer can be discerned. This difference is to be expected, due to the difference in orientation of the silica fibers with respect to the engine axis between the nozzle and chamber liner. The rate of heat conduction apparently was greater in the chamber than in the nozzles as evidenced by the difference in char depths.

Total usage time for the nozzles was limited by a allowable throat diameter growth of about 10% and amounted to approximately 2 seconds. Chamber liners were used up to 5 seconds total time which resulted in a 30 to 50% loss of liner material.

IV. DISCUSSION OF RESULTS

The data were studied in an attempt to find a correlation between the ablation rate and the various measured parameters as part of an effort to correlate ablation rate and heat transfer. Four different approaches were used in studying the data. These approaches are discussed subsequently and will be referred to as the inspection method, the linear coefficient method, the correlation method, and the nondimensionalization method.

1. Inspection Method. This method simply consisted of an organized parametric grouping of run data sets. The data were grouped such that corresponding data in a given group had similar values, within limits, except for one variable (e.g. mixture ratio or chamber pressure). In this manner it was hoped to study the individual effect of each data parameter upon the ablation rate, however, due to the wide variation in run conditions obtained no useful information was acquired with this method.

2. Linear Coefficient Method. In this method systems of simultaneous linear equations were formed from the run data by arbitrarily selecting a group of run data sets, the number of sets in the group being equal to the number of parameters being considered. The parameters in each of these data sets were then multiplied by a given set of unknowns, the resultant products were summed by data set, and the sums were equated to the linear ablation rate reported in each set. The objective here was to solve a large number of cases and obtain a representative set of unknowns for all the data that could be used to predict ablation rate. The solutions, however, from a large number of cases revealed no representative set of solution numbers.

3. Correlation Method. In this method the ablation data was input to a statistical data correlation program (21). This program computed correlation coefficients relating all the parameters to the ablation rate. The computed correlation coefficients had values, for the most part, of less than .5.

4. Nondimensionalization Method. The ablation rate was considered to be a function of run duration, chamber pressure, local mass flux, and throat diameter. Application of the Buckingham Pi Theorem yielded two nondimensional parameters. The data were plotted using the two dimensional variables as coordinates. No trends were found in the plot to indicate the effect of the data parameters upon the ablation rate.

The primary reason for the lack of correlation among the ablation data is thought to be the short durations of the engine firings. The actual length of steady state engine operation for most firings was .2 to .3 seconds accounting for only one-third to one-half of the total test length as set on sequence timers. Therefore, startup and shutdown transients had a significant effect upon the performance of the ablative material.

V. SUMMARY OF RESULTS

Ablation cooling was successfully used to cool a liquid bipropellant rocket engine at a nominal chamber pressure of 4000 psia (272.2 atm). The ablative material used was Western Backing 2230, a composite of silica cloth and phenol aldehyde resin. The average overall linear ablation rate obtained at the nozzle throat was 0.0861 in/sec (0.219 cm/sec). Ablative nozzle life time for the material was limited by an allowable throat diameter growth of about 10% and was approximately 2 seconds. Ablative chamber liners would accomodate about 5 seconds total usage. One linear ablation rate measurement was performed on a chamber liner at the end next to the nozzle and was found to be 0.071 in/sec (0.180 cm/sec). The orientation of the silica fibers, which was parallel to the engine axis in the chamber liners and perpendicular to the axis in the nozzles, had a pronounced effect on the depth of the char layer obtained.

BIBLIOGRAPHY

1. Adams, Mac C., "Recent Advances in Ablation", A. R. S. Journal, Volume 29, Number 9, Sept. 1959, pp. 625-632.

2. Swann, Robert F., "Approximate Analysis of the Performance of Char-Forming Ablators", NASA TR R-195, June 1964.

3. Rivers, W. J.; Van Wyk, R.; Seader, J. D.; Friedman, H. A.; Chu, H. N.; "Effect of Rocket Engine Combustion on Chamber Materials, Part I: One-Dimensional Computer Program", AFRPL-TR-65-13, January, 1965.

4. Swann, Robert F.; Pittman, Claud M.; Smith, James C.; "One-Dimensional Numerical Analysis of the Transient Response of Thermal Protection Systems", NASA TN D-2976, September, 1965.

5. McFarland, B. F. and Joerg, P., "A Theory for the Ablation of Heterogeneous Materials"; Aerojet-General Corporation, prepared for presentation at the ChE National Meeting in New Orleans, March 10, 1963.

6. Campbell, J. G.; Carey, M. D.; Sanders, C. F., Jr.; Coulbert, C. D.; "Spacecraft Engine Thrust Chamber Cooling", Marquardt Corporation Report 6069, July, 1964.

7. Rindal, R. A., "A Numerical Solution for Heat Conduction in a Slab Characterized by Two Receding Surfaces", Thiokol Chemical Corporation Technical Note 53/C-TN-28, January 10, 1962.

8. Curry, Donald M., "An Analysis of a Charring Ablation Thermal Protection System", NASA TN D-3150, 1965.

9. Scala, Sinclaire M. and Gilbert, Leon M., "Thermal Degradation of a Char-Forming Plastic During Hypersonic Flight", ARS Journal, Volume 32, Number 6, June 1962, pp. 917-924.

10. Kauzlarich, J. J., "Ablation of Reinforced Plastic for Heat Protection", J. of Applied Mechanics, Volume 32, Series E, Number 1, March, 1965, pp. 177-182.

11. Brandon, George W., Jr., "Heat-Shield Ablation Measurements Using Radioisotope Techniques", NASA TN D-3329, March 1966.

12. Skinn, Authur M., Jr., "Experimental Evaluation of Six Ablative-Material Thrust Chambers as Components of Storable-Propellant Rocket Engines", NASA TN D-3945, June, 1967.

13. McLain, Allen G.; Sutton, Kenneth; Walberg, Gerald D.; "Experimental and Theoretical Investigation of the Ablative Performance of Five Phenolic-Nylon-Bond Materials", NASA TN D-4374, April, 1968.

14. Davis, William G., "Determination of Heat-Shield Char-Front Recession with a Nucleonic Technique", NASA TN D-3264, March, 1966.

15. Bartz, D. R., "Survey of the Relationship between Theory and Experiment for Convective Heat Transfer from Rocket Combustion Gases", condensed version of a chapter in Advances in Heat Transfer, Vol. II, edited by J. P. Hartnett and T. F. Irvine, Jr., Academic Press. Inc., 1965.

16. Dow, Marvin B. and Swann, Robert F., "Determination of Effects of Oxidation on Performance of Charring Ablators", NASA TR R-196, June, 1964.

17. Tompkins, Stephen S. and Dow, Marvin B., "Parametric Study of a Low-Density Charring Ablator", NASA TN D-3735, Dec., 1966.

18. Walberg, Gerald D., "Analytical Study of Diffusion-Controlled Char Oxidation and Its Effect on Steady-State Ablation of Plastic Materials", NASA TR R-242, July, 1966.

19. Welsh, William E., Jr., and Witte, Arvel B., "A Comparison of Analytical and Experimental Local Heat Fluxes in Liquid-Propellant Rocket Thrust Chambers", ASME Journal of Heat Transfer, Vol. 20, Feb., 1962.

20. Hines, W. S.; Combs. L. P.; Ford, W. M.; Van Wyk, R.; "Development of Injector Chamber Compatibility Analysis", AFRPL-TR-70-12, March, 1970.

21. Health Sciences Computing Facility, UCLA, Biomedical Computer Programs, "Correlation with Transgeneration", BMD2D, September, 1965.

22. McBride, Bonnie and Gordon, Sanford, ICRPG, "One-Dimensional Equilibrium Reference Program", May, 1968.

NO. OF FIRINGS ON NOZZLE	CORRESPONDING RUN NOS.	RUN DURATION (SEC)	CHAMBER PRESSURE (PSIA)	MIXTURE RATIO	INJECTOR TUBE DIAMETERS (INCHES)	PULSE	ABLATION RATE (IN/SEC)
2	7,8	0.293	3855.	1.42	OX-0.118 FUEL-0.106	NO	0.0598
3	7, 8, 9	0.301	3955.	2.58			0.1129
4	7,8,9,10	0.460	4015.	1.87			0.1200
1	11	0.343	3915.	2.42			0.0728
1	12	0.430	3995.	2.58			0.0858
2	12, 13	0.356	3915.	1.26			0.1053
3	12, 13, 14	0.329	4045.	1.54			0.0400
1	15	0.302	4005.	2.20			0.0811
2	15, 16	0.523	4025.	2.07	OX-0.094 FUEL-0.075		0.1050
1	18	0.527	3725.	1.81			0.1723
1	19	0.490	3995.	1.98			0.1112
2	19, 20	0.536	3865.	2.00			0.0755
1	21	0.546	4040.	1.50			0.0694

TABLE I. NOZZLE THROAT ABLATION DATA

NO. OF FIRINGS ON NOZZLE	CORRESPONDING RUN NOS.	RUN DURATION (SEC)	CHAMBER PRESSURE (PSIA)	MIXTURE RATIO	INJECTOR TUBE DIAMETERS (INCHES)	PULSE	ABLATION RATE (IN/SEC)
2	21, 22	0.637	3765.	1.70	OX-0.094 FUEL-0.075	NO	0.0942
1	23	0.422	3965.	2.27			0.0805
2	23, 24	0.455	3945.	1.94			0.1367
3	23, 24, 26	0.422	3850.	1.74			0.0498
4	23,24,26,27	0.430	3800.	2.00			0.0384
1	28	0.215	3868.	2.31			0.1653
2	28, 29	0.365	3927.	2.14			0.1108
3	28, 29, 30	0.560	3801.	1.89			0.0837
1	31	0.370	3913.	2.26			0.0881
2	31, 32	0.405	4067.	1.08			0.0567
2	34, 35	0.577	3910.	1.88	OX-0.118 FUEL-0.106		0.0530
1	36	0.616	4090.	1.45		YES	0.0386
2	36, 37	0.527	4060.	1.76			0.0775

TABLE I., CONT. NOZZLE THROAT ABLATION DATA

NO. OF FIRINGS ON NOZZLE	CORRESPONDING RUN NOS.	RUN DURATION (SEC)	CHAMBER PRESSURE (PSIA)	MIXTURE RATIO	INJECTOR TUBE DIAMETERS (INCHES)	PULSE	ABLATION RATE (IN/SEC)
1	38	0.593	4020.	1.84	OX-0.118 FUEL-0.106	NO	0.0580
1	41	0.520	3800.	2.67		YES	0.0540
3	41, 42, 43	0.370	4060.	2.99			0.0850
4	41,42,43,44	0.430	4100.	2.25			0.1850
1	45	0.440	4100.	2.10			0.0535
2	45, 46	0.640	4120.	1.42			0.0703
1	47	0.390	4080.	1.73		NO	0.0705
2	47, 48	0.410	4070.	1.89			0.0817
1	49	0.385	3800.	2.15	OX-0.094 FUEL-0.075		0.0675
2	49, 50	0.520	3900.	1.53			0.0904

TABLE I., CONT. NOZZLE THROAT ABLATION DATA

NO. OF FIRINGS ON NOZZLE	CORRESPONDING RUN NOS.	RUN DURATION (SEC)	CHAMBER PRESSURE (ATM)	MIXTURE RATIO	INJECTOR TUBE DIAMETERS (CM)	PULSE	ABLATION RATE (CM/SEC)
2	7, 8	0.293	262.3	1.42	OX-0.300 FUEL-0.269	NO	0.1519
3	7, 8, 9	0.301	269.1	2.58			0.2868
4	7,8,9,10	0.460	273.2	1.87			0.3048
1	11	0.343	266.4	2.42			0.1849
1	12	0.430	271.8	2.58			0.2179
2	12, 13	0.356	266.4	1.26			0.2675
3	12,13,14	0.329	275.2	1.54			0.1016
1	15	0.302	272.5	2.20			0.2060
2	15, 16	0.523	273.9	2.07			0.2667
1	18	0.527	253.5	1.81	OX-0.239 FUEL-0.191		0.4376
1	19	0.490	271.8	1.98			0.2824
2	19, 20	0.536	263.0	2.00			0.1918
1	21	0.546	274.9	1.50			0.1763

TABLE I., CONT.   NOZZLE THROAT ABLATION DATA

12

NO. OF FIRINGS ON NOZZLE	CORRESPONDING RUN NOS.	RUN DURATION (SEC)	CHAMBER PRESSURE (ATM)	MIXTURE RATIO	INJECTOR TUBE DIAMETERS (CM)	PULSE	ABLATION RATE (CM/SEC)
2	21, 22	0.637	256.2	1.70	OX-0.239 FUEL-0.191	NO	0.2393
1	23	0.422	269.8	2.27			0.2045
2	23, 24	0.455	268.4	1.94			0.3472
3	23,24,26	0.422	262.0	1.74			0.1265
4	23,24,26,27	0.430	258.6	2.00			0.0975
1	28	0.215	263.2	2.31			0.4199
2	28, 29	0.365	267.2	2.14			0.2814
3	28,29,30	0.560	258.6	1.89			0.2126
1	31	0.370	266.3	2.26			0.2238
2	31, 32	0.405	276.7	1.08			0.1440
2	34, 35	0.577	266.1	1.88	OX-0.300 FUEL-0.269		0.1346
1	36	0.616	278.3	1.45		YES	0.0980
2	36, 37	0.527	276.3	1.76			0.1969

TABLE I., CONT. NOZZLE THROAT ABLATION DATA

NO. OF FIRINGS ON NOZZLE	CORRESPONDING RUN NOS.	RUN DURATION (SEC)	CHAMBER PRESSURE (ATM)	MIXTURE RATIO	INJECTOR TUBE DIAMETERS (CM)	PULSE	ABLATION RATE (CM/SEC)
1	38	0.593	273.5	1.84	OX-0.300 FUEL-0.269	NO	0.1473
1	41	0.520	258.6	2.67		YES	0.1372
3	41, 42, 43	0.370	276.3	2.99			0.2159
4	41,42,43,44	0.430	279.0	2.25			0.4699
1	45	0.440	279.0	2.10			0.1359
2	45, 46	0.640	280.3	1.42			0.1786
1	47	0.390	277.6	1.73		NO	0.1791
2	47, 48	0.410	276.9	1.89			0.2075
1	49	0.385	258.6	2.15	OX-0.239 FUEL-0.191		0.1715
2	49,50	0.520	265.4	1.53			0.2296

TABLE I., CONT. NOZZLE THROAT ABLATION DATA

NOZZLE DIVERGENT SECTION

NOZZLE HUB

ABLATIVE RING SEAL

SEAL RING

NOZZLE

THRUST CHAMBER LINER

THRUST CHAMBER

INJECTOR

4.410 DIA.

3.85

.90 DIA.

4.000

2.340 DIA.

8.425
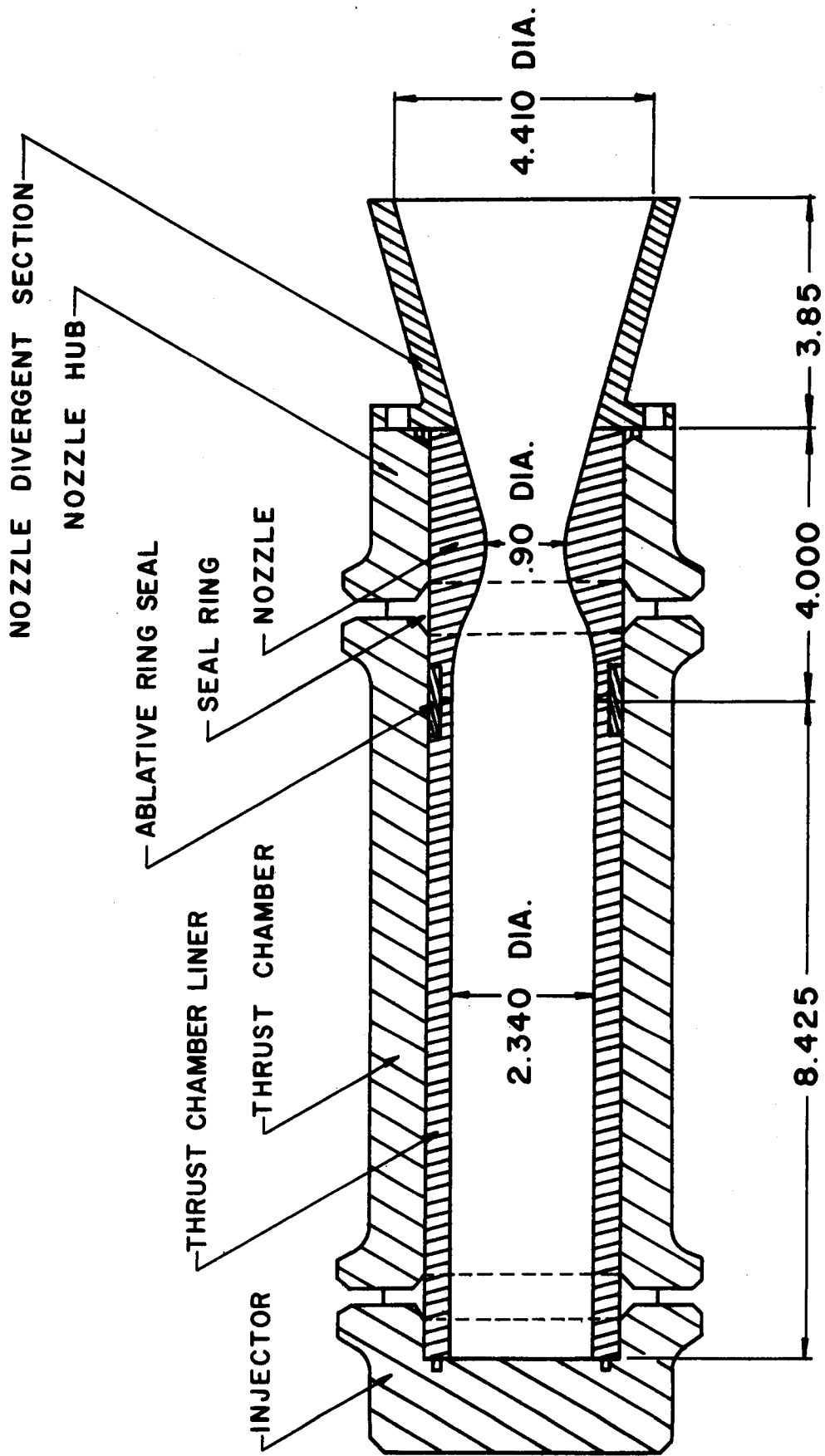
FIGURE 1. CROSS-SECTION SKETCH OF ROCKET ENGINE

15

1.5937 DIA. BSC

A

A

6.000 DIA.

3.250 DIA.

2.340 DIA.

INJECTOR

ABLATIVE CHAMBER LINER

2.427

SECTION A-A

.500 DIA. BSC

FUEL- 0.134 DIA. OR 0.109 DIA.

OXIDIZER- 0.148 DIA. OR 0.119 DIA.

B

B

DETAIL OF QUADLET

45°

45°
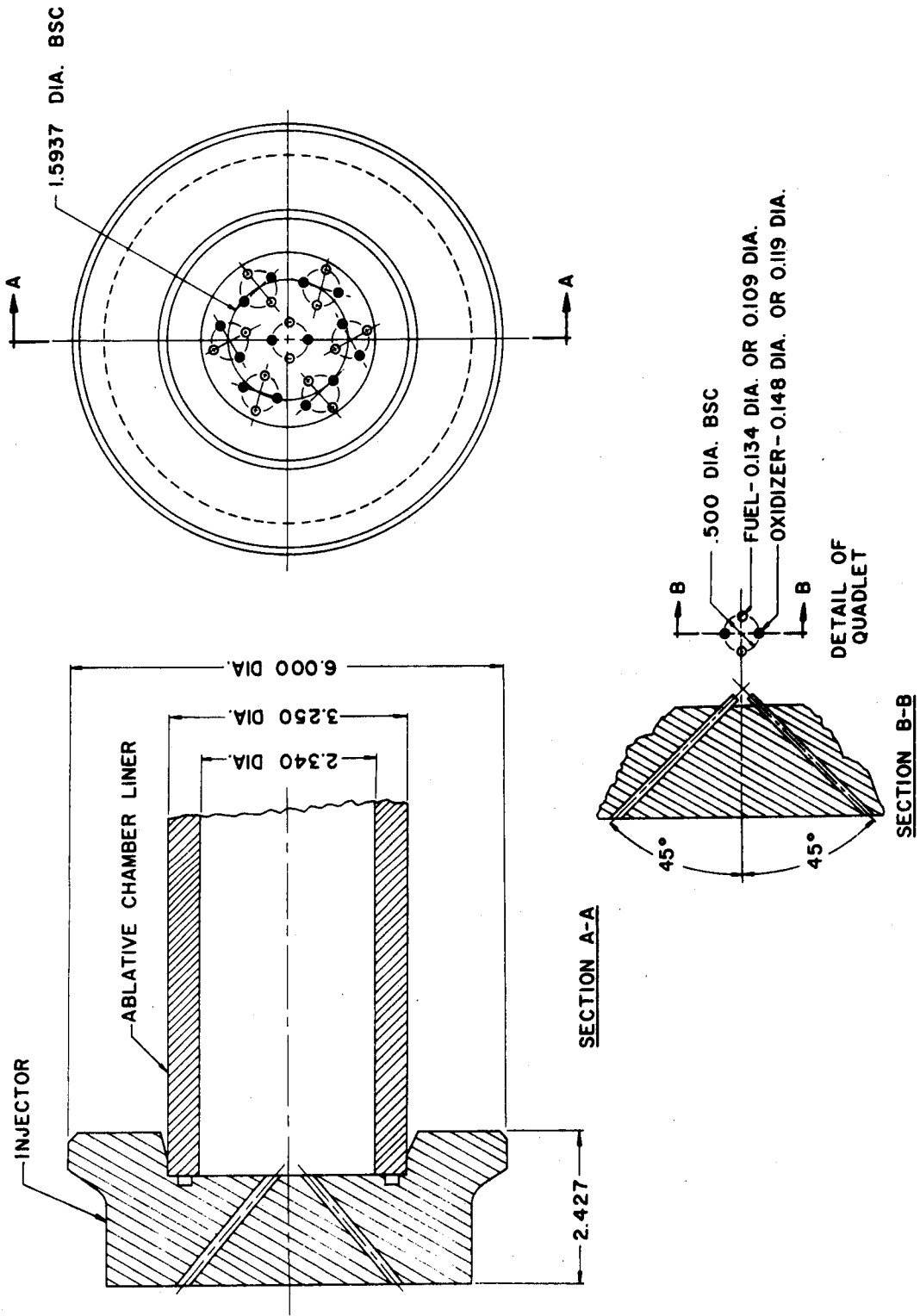
SECTION B-B

FIGURE 2. INJECTOR DRAWING

16

OUTLINE OF ABLATED NOZZLE
THROAT (ENLARGED)
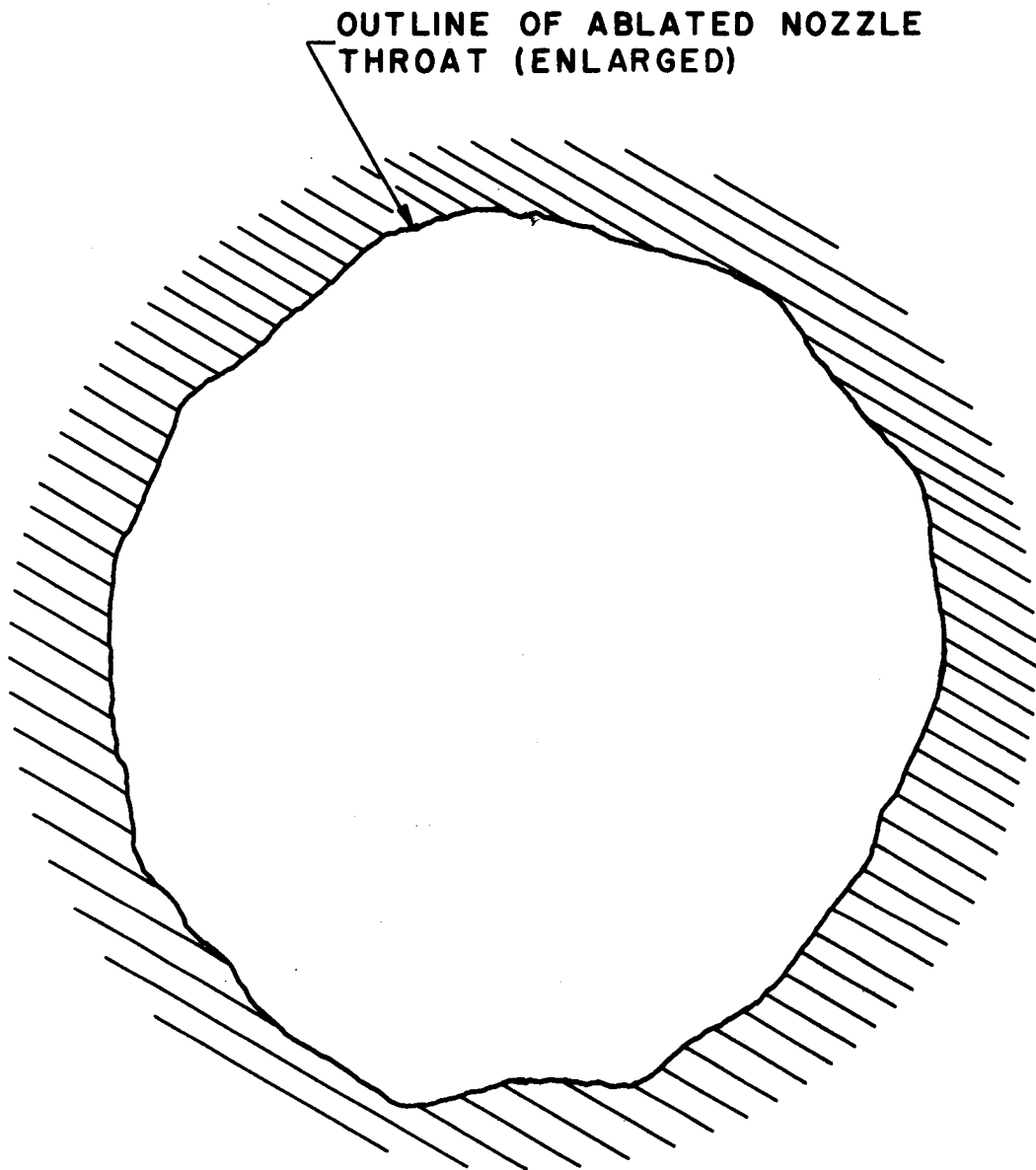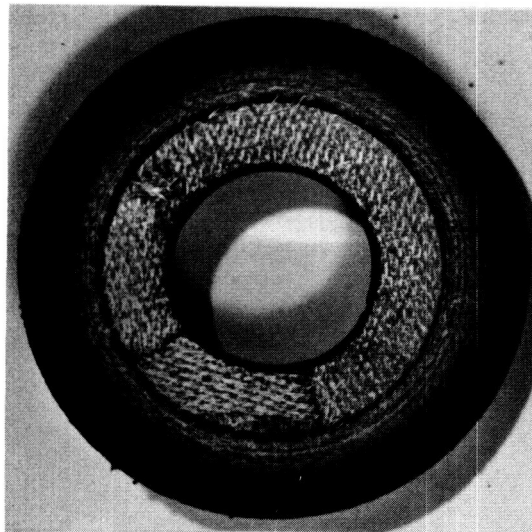
FIGURE 3.   TYPICAL CROSS-SECTION OF
ABLATED   NOZZLE   THROAT

17
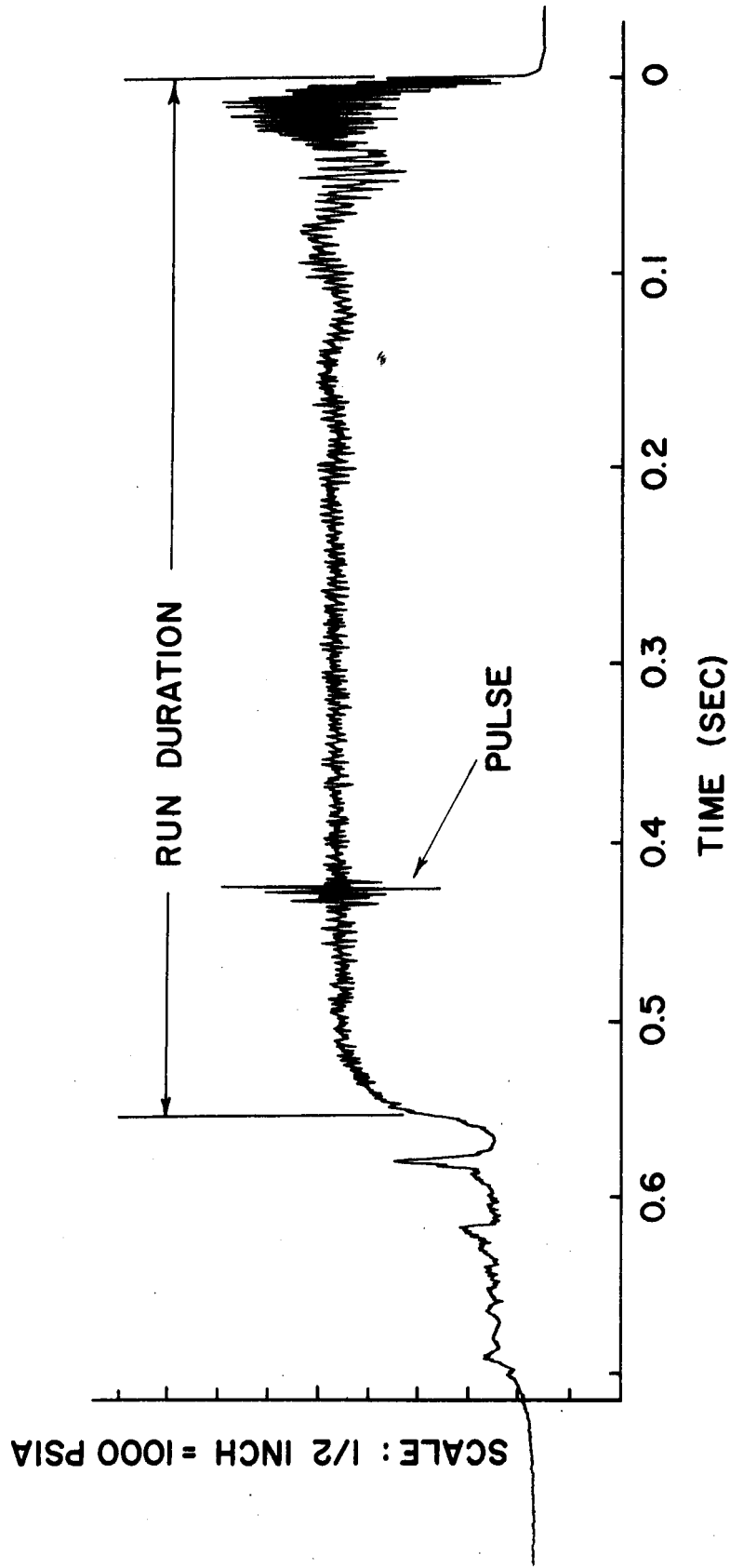
FIGURE 4. OSCILLOGRAPH TRACE OF CHAMBER PRESSURE DURING ENGINE FIRING
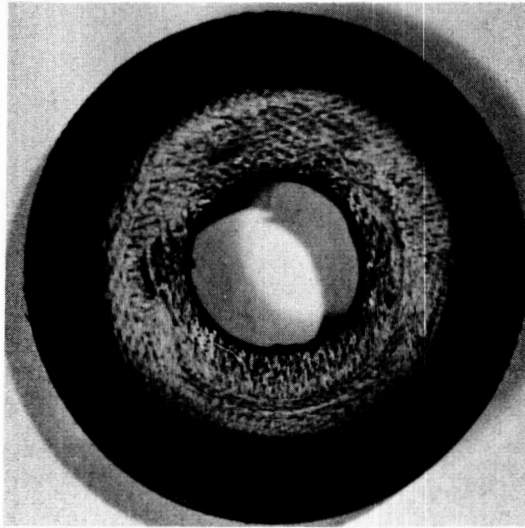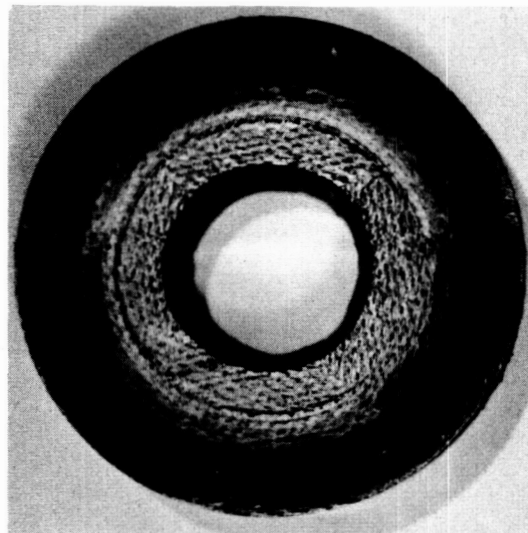
18
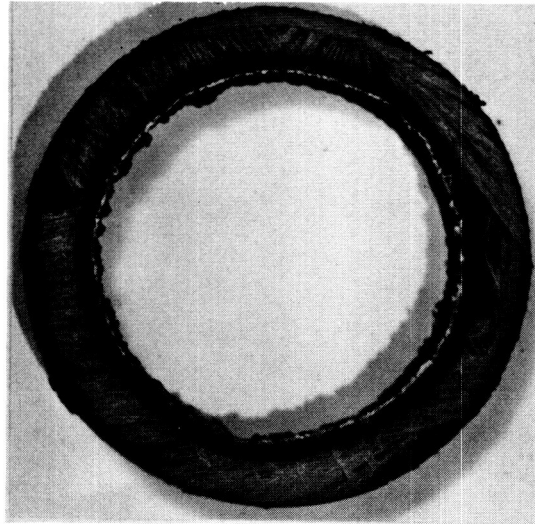
5a. CHAMBER LINER

5b. NOZZLE

FIGURE 5. CROSS SECTION OF
ABLATED MATERIAL

5c. NOZZLE

5d. NOZZLE

FIGURE 5, CONT. CROSS SECTION OF
ABLATED MATERIAL