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Technical Report No. 32-978

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PREFACE

The test program described in this Report was directed toward establishing design limits for state-of-the-art ablative thrust chambers. The standard test conditions were purposely made especially severe on a one-shot, no modification basis. Thus the tests were not intended as an evaluation of vendor capability nor should they be so construed. Certainly many design and material improvements have been made since these designs were finalized early in 1964.

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ABSTRACT

Five 100-lb-thrust ablatively cooled chambers, designed and fabricated by industrial companies, were subjected to test firings in a partial assessment of the state of the art of this type of chamber design. The severity of the test conditions (nitrogen tetroxide and hydrazine burned at a nominal pressure of 150 psia) caused all but two of the chambers to fail or erode excessively before 500 sec of firing time had elapsed. Both of the more resistant chambers had hard throat inserts of silicon carbide or molybdenum.

I. INTRODUCTION

This Report describes several thrust chambers designed for liquid-bipropellant rocket engine operation and the results obtained when these chambers were test fired. The task of procuring, testing, and evaluating the chambers was accomplished during 1964 and early in 1965 as part of the Laboratory's Advanced Liquid Propulsion Systems (ALPS) program.

A. Purpose

The objective of this effort was to partially assess the state of the art of designing and fabricating small, ablatively cooled thrust chambers such as might be supplied by industry for application in spacecraft.

B. Method

Several industrial companies with ablative-chamber engineering design capability were invited to propose an optimum design to meet the requirements of a procurement specification. The specification (see the Appendix) was made as general as possible to allow each supplier sufficient latitude to design the best chamber his capabilities permitted. Materials selection and processing procedures were left completely to the discretion of the suppliers. Lenient limitations were put on the weight and the outside envelope. Only the internal geometry and attachment-flange bolt circle were firmly controlled; these parameters had to be fixed to assure comparability of the test results and compatibility of the chambers with the injector to be used in the test firings.

Test conditions were specified so that the chamber designer would take these into consideration. These conditions were a reasonable compromise between a meaningful approximation of future propulsion needs and test convenience. Exact propulsion-system details for future spacecraft were not known, but it was assumed that, at least for several missions planned for the near future, conservatism would dictate the use of "Earth storable" propellants (see Ref. 1 for a discussion of the reasons for this conclusion). Nitrogen tetroxide (N_2O_4) and hydrazine (N_2H_4) were specified because this was the propellant combination originally selected for the ALPS program and an existing fully characterized injector was available. The thrust level (100 lb) was also chosen to match the injector, although it was realized that this thrust may be somewhat low for certain spacecraft. To enable them to deliver high total impulses, small low-thrust engines must be capable of very long burning times. The test firing was to continue uninterrupted until prescribed limits on external temperature (400°F) or thrust-vector degradation (allowable side-force variation was limited to 5% of the axial thrust) were exceeded. (No restart or vacuum testing was planned, but it was stated that the ability of the chamber to be stored and repeatedly refired in the space environment would be estimated as part of the evaluation.) It was predicted that the test injector would produce a characteristic velocity of 5450 to 5550 ft/sec at a chamber pressure of 150 psia. This performance level was about as good as the best injectors then in use with ablative chambers, but the pressure was considerably above what is commonly used, which is in the range of 85 to 125 psia.

C. Discussion

Despite the current limitations on the conditions of their usage (burning time, combustion temperature, and chamber pressure), ablative chambers are popular because of the simplicity of their construction and the ease with which they can be integrated into a propulsion system. These attributes are primarily a result of their built-in cooling potential, which obviates regenerative cooling. Eliminating the regenerative circuit reduces propellant-pressure losses, decreases the likelihood of propellant decomposition, removes the chance of freezing propellant in the passages during coast in space, and improves the transient-response capabilities of engines. While possessing these advantages over regeneratively cooled chambers, the ablative chamber may also have characteristics that allow it to compete favorably with radiation-cooled chambers. For example, the relatively cool outer wall allows the ablative chamber to be installed within recessed locations where the heat radiating from hot walls might be intolerable. In addition, because of their cooler operating wall temperatures, ablative chambers may be easily attached to injectors; this is a problem area with radiation-cooled chambers because of their much higher wall temperatures.

Even with these advantages, however, ablative chambers are not automatically the proper choice for all applications. The factors that make it difficult to predict how well such chambers can be made to perform are that the real conditions in operating chambers are not known exactly and the analytical models describing the theoretical behavior are very complex; also, because of the number of different materials now available and because the number of design geometries is infinite, the matrix of possible designs is very large.

Presently available empirical data do not provide many answers either, since most of the data are not comparable. Some data are for nozzle sections only, while others are for combustion-chamber sections or complete thrust chambers. Even when limiting a survey to nozzle sections, not all data in the literature can be correlated because some investigators use water-cooled chambers upstream of the nozzle, thus creating a cooler-than-normal boundary layer, while others use ablating materials upstream to gain more realistic operating conditions. Other parameters such as the propellant combination, chamber pressure, mixture ratio, and performance level also vary widely from one evaluation to the next. These differences make correlation between programs almost hopeless.

Interpreting the data produced by individual programs may also be difficult because the test conditions are not precisely known. If the erosion rates and temperature distributions are actually functions of local heat flux, local shear forces in the boundary layer, and the local chemical composition of the gases, then these parameters must be known for each test or they must be held constant from test to test in order to make the data meaningful. The effects of the variations in the values of these parameters must be known to permit extrapolation to untested conditions. If testing is performed at the actual flight conditions, then repeatability of the conditions must be obtained to assure comparability of the data; this would be necessary both for screening materials and designs and for checking the reproducibility (quality control) of the chosen flight-chamber design.

The ideal way to evaluate a test program would be to know the exact conditions in the chamber during the test, but at the present state of the art in rocketry it is not generally feasible to precisely measure the local conditions in a combustion chamber. The best alternative available to the experimenter, then, is to keep conditions constant from test to test. Fortunately, there is a reasonably easy means to control the conditions in the chamber.

It is a widely accepted opinion that, for a given chamber geometry, the injector controls the combustion conditions. By this it is meant that the time-average distribution of mass flux, mixture ratio, and heat flux throughout the chamber and nozzle converging section is determined by the injector geometry and the hydraulic conditions resulting from the injector geometry. The injector may be of such a design as to provide the same combustion conditions time after time, or it may be marginal in certain aspects so that unpredictable changes in conditions may occur from test to test.

Unintentional variability in an injector may result from a number of causes. Poor manifolding may cause unevenness in the distribution of the propellants among the orifices. Poor orifice entry or exit conditions, such as high cross velocities, sharp edges, or burrs, may cause cavitation or randomly skewed velocity profiles in the liquid jets, sheets, or sprays. Short orifices (low L/D values) make the properties of the emerging liquid jet particularly prone to anomalies caused by the upstream conditions in the manifolds and orifice entries; the jets, sheets, or sprays may be randomly flickering, misdirected, or broken up into a spray of large droplets. Any of these deviations from optimum hydraulic injection and impingement may cause localized conditions in the boundary layer or on the ablative surface that increase the local ablation rate in an unreproducible manner.

To assure comparability of results, the series of evaluation firings reported herein was undertaken with special emphasis on controlled, reproducible injection and thus reproducible combustion. Section II describes the equipment and procedures used to secure this reproducibility.

II. TEST EQUIPMENT AND PROCEDURE

In order to achieve reproducible test conditions, it was necessary to examine the fundamental sources of inconsistencies and then to choose test equipment and procedures that would eliminate or minimize the effect of these inconsistencies during the testing.

The most crucial choice, that of the injector, was resolved by the selection of a well-proven design that incorporated a number of features intended to minimize the variations (with time) of local mass and mixture ratio. Total propellant-flow rate was made nearly independent of chamber pressure by using a high-pressure-loss feed system so that the controlling resistance to flow was the frictional loss in the plumbing, not the back-pressure in the chamber. Differences in ambient temperature and wind conditions may have caused some variations of heat transfer due to free convection. (External free-convection cooling could have been reduced to insignificance had a vacuum facility capable of a sufficiently long burning time been available.) Conduction-cooling by the injector was somewhat variable because of differences in the chamber designs, but all chambers were attached to the test injector in the same manner with essentially the same bearing pressure. This was accomplished by means of eight bolts or studs on which the load was kept fairly constant, regardless of thermal expansion of the injector or chamber flange, by compressed springs between the nuts and the injector flange. The joint between the injector and chamber flanges was sealed by 1/32-in.-thick flat gaskets of V-44 uncured rubber[1], the rubber gasket acting as both a thermal insulator and chamber-pressure seal. Differences in radiation-heat exchange with the surroundings would be caused solely by the differences in the chambers themselves, since the test setup remained constant from test to test. Vibration levels were insignificant because of the smoothness of the combustion process associated with the test injector.

All of the deterioration in the chambers was attributed to the test firings (although it is possible that the process of attaching the thermocouples could have had a minor effect upon some of the materials). No liquids, which might soak into the chamber walls and then expand when heated, were allowed to contact the chambers before the test. All leak tests were made with dry nitrogen gas so that any absorbed fluid could outgas readily.

Each chamber was subjected to only one test firing so that cumulative damage from repeated cycles of heating up and cooling down would not obscure the

[1]General Tire and Rubber Company, Akron, Ohio.

effects caused by the firing environment. The firing sequence, with the exception of the burning time, was made as nearly alike from test to test as possible. Start and shutdown transients were quick and very smooth. The time from the first sign of chamber-pressure rise to full pressure was typically 300 msec. A shutdown purge of nitrogen gas helped to scavenge the excess oxidizer from the chamber at the end of each test.

A. Injector

The JPL Mod IV 100-lb-thrust injector was chosen for these tests because of its demonstrated reproducibility from firing to firing. This injector was designed to operate at a nominal chamber pressure of 150 psia and a thrust level of 100 lbf when injecting nitrogen tetroxide–hydrazine propellants at a mixture ratio of 1.2. It has 10 identical unlike-impinging-doublet elements arranged so that the injected mass is spread across the injector face in a relatively uniform manner as shown in Fig. 1. Six of the elements are arranged in a circle around the circumference of the injector face. The remaining four elements are arbitrarily placed in the center of this circle. The wall of a thrust chamber was thus presented with a relatively uniform flow produced by the six "boundary" elements. For the tests reported here, the injector was manifolded so that the portion of the spray fan closest to the combustion-chamber wall constituted the lowest-mixture-ratio or fuel-rich zone (assuming that the propellant streams penetrate each other—see Ref. 2).

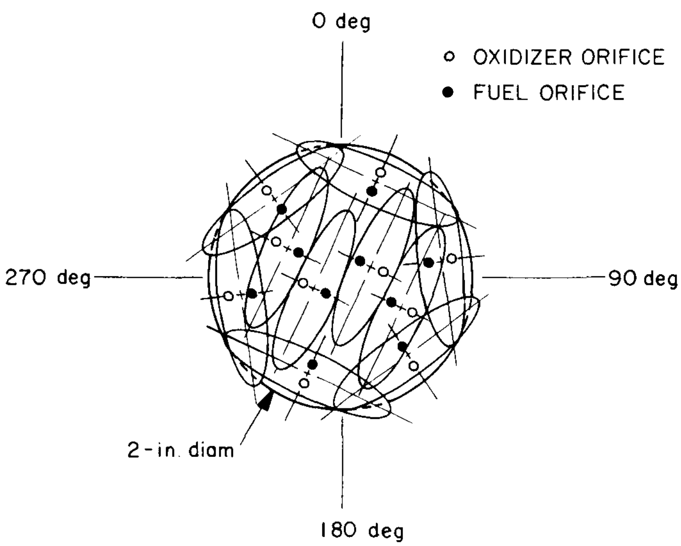

Fig. 1. Arrangement of the 10 pairs of propellant orifices and resulting sprays from the Mod IV 100-lb-thrust injector

The most notable design feature of this injector is the use of long orifices (having a length-to-diameter ratio L/D of 100 and a nominal internal diameter of 0.020 in.), through which the propellants are injected. These long tubes were designed to provide reproducible fuel and oxidizer jets and, hence, reproducible spray properties within the combustion chamber. At relatively high Reynolds numbers ($N_{Re} > 10^4$), this orifice configuration produces a liquid jet that is characterized by a fully developed turbulent-flow velocity profile, which is relatively insensitive to hydraulic disturbances in the upstream feed system.

Two equal-diameter orifices create a doublet element meeting Rupe's optimum mixing criteria (Ref. 3) with N_2O_4–N_2H_4 at a mixture ratio of 1.2 if the velocity ratio of the jets is 0.83 (oxidizer/fuel). In the Mod IV injector, the rated oxidizer-jet velocity is 150 ft/sec at a differential pressure of 440 psi across the oxidizer orifices and the rated fuel velocity is 180 ft/sec at 460 psi across the fuel orifices. These jets are impinged at an included angle of 60 deg to produce a spray composed of very finely atomized droplets.

Calibration firings in a heavy, uncooled metal thrust chamber with a characteristic length L^* of 42 in. indicated that this injector produced a characteristic velocity c^* of 5490 ft/sec $\pm$ 1% at nominal conditions (N_2O_4–N_2H_4 at r = 1.2 and P_c = 150 psia). The normal operating sequence attained steady-state propellant-flow rates and chamber pressure within 300 msec of ignition. Ignition and steady-state operation are extremely smooth; maximum peak-to-peak chamber-pressure variations during steady state are only $\pm$2 psi (measured with a flush-mounted Photocon pressure transducer)[2] and the recorded wave form appears to show a random, noiselike frequency distribution. Further information on the performance of this injector is contained in Ref. 4.

The circumferential distribution of heat flux to the combustion-chamber wall was determined during short-duration uncooled-steel-chamber tests by using the transient thermocouple-plug technique discussed in Ref. 5. During these tests, the engine operating conditions (chamber pressure, mixture ratio, and thrust-chamber geometry) were maintained as closely as possible to the conditions in the ablative-chamber tests. The circumferential distributions of chamber heat flux were obtained from three thermocouples located 45 deg apart on the chamber wall at an axial location 1.0 in. upstream of the nozzle inlet. As seen in Fig. 2, the circumferential

[2]Model 307, Photocon Research Products, Pasadena, Calif.

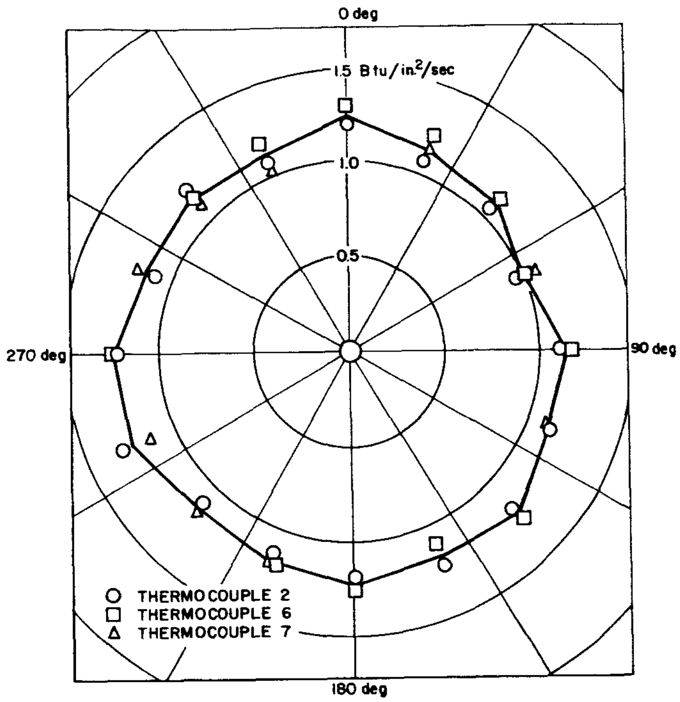

Fig. 2. Circumferential distribution of heat flux produced by the Mod IV injector

local heat flux was found to be fairly uniform at a time when the chamber-wall temperature was approximately 500°F. In addition, circumferential heat-flux distributions were obtained at stations 0.67 in. downstream and 0.67 in. upstream of the 1.0-in. location. These data show similar local magnitudes of heat flux, indicating that extreme local chamber-wall hot spots are unlikely with this injector.

The stainless-steel injector face was internally water-cooled to prevent overheating during long-duration firings. The outer edge of this cooled face serves as the flange to which the chambers are bolted and by which the complete injector-chamber assembly is clamped into the thrust mount.

B. Test Procedure

All of the evaluation firings were made on "D" Stand at JPL's Edwards Test Station. Each chamber was fired uninterruptedly until (1) the chamber pressure dropped below 120 psia, (2) the outside-wall temperature exceeded 400°F, or (3) an extraordinary change in axial or side thrust occurred. The firing tests were made at local ambient temperature and pressure (atmospheric pressure approximately 13.7 psia) since the available altitude-test facility was limited to operating periods of about 200 sec.

Propellants were fed to the engine from remote tanks pressurized with nitrogen gas to between 650 and 740 psia. As mentioned previously, the very high frictional losses in the feed system resulted in nearly constant propellant-flow rates regardless of the decay in chamber pressure due to chamber-throat-area change. Purges of gaseous nitrogen were forced through the injector orifices before and after firing, primarily to prevent the buildup of deposits on the face of the injector or plugging of the orifices; there was concern that such deposits might result from volatile substances outgassing from the hot ablative materials and condensing on the cooler injector surfaces.

Measurements taken and recorded during the firings included thrust forces, pressures, flows, and temperatures. Use of a six-component thrust-measuring system made it possible to record changes in axial thrust, roll (about the axial thrust), and pitch and yaw forces due to changes in the inner contour of the ablative chambers. Figure 3 shows the thrust-measuring system; additional information about this equipment can be found in Ref. 6. Chamber pressure was sensed by a Taber transducer[3] attached to a tap through the face of the injector. Propellant-flow rates were measured by turbine-type meters.[4] Twelve thermocouples were cemented[5] to the external surface of each chamber to provide continuous data on the surface temperatures during and after the firing test. Four thermocouples were mounted along each of three different longitudinal lines, as shown in Fig. 4, so as to get data on both the axial and circumferential distribution of temperatures. The circumferential locations were chosen to be representative of the conditions adjacent to three different propellant-mass-flux densities (based on the distribution produced by the element arrangement shown in Fig. 1). In addition to this instrumentation, motion picture cameras were used to make color photographs of each test to supply visual evidence of the behavior of the chamber.

Since each test was to be terminated when certain "red-line" values were exceeded, most of the measured parameters were monitored in real time by the test personnel. Also, the chamber under test was viewed by means of a closed-circuit television system so that leaks, incipient structural failures, and other obvious justifications for a test termination could be detected.

[3]Taber Instrument Corp., North Tonawanda, N. Y.
[4]Fischer and Porter Company, Warminster, Pa.
[5]The thermocouples were attached by JPL with a refractory cement cured according to the recommendations of the chamber supplier.

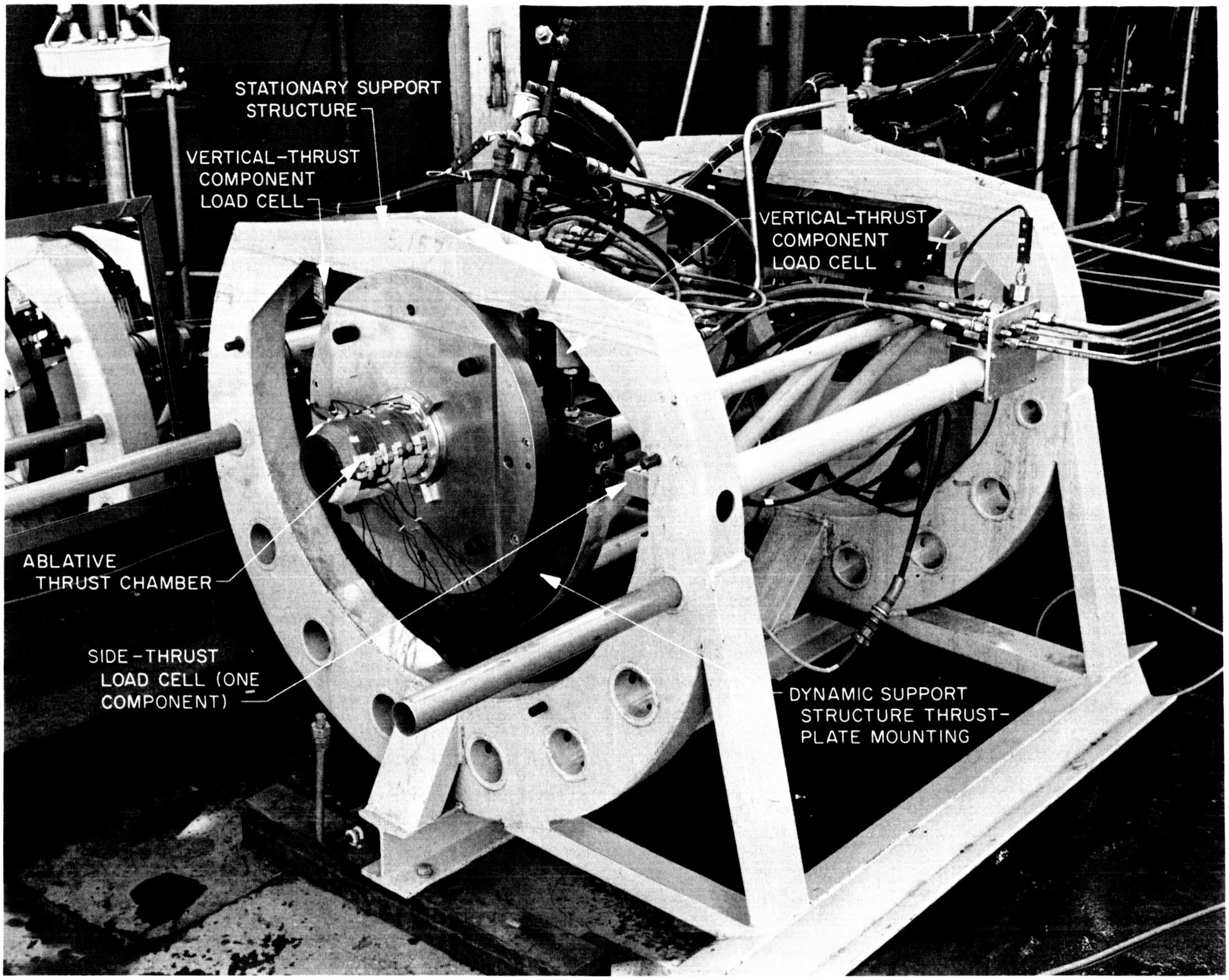

Fig. 3. Six-component thrust-measuring system

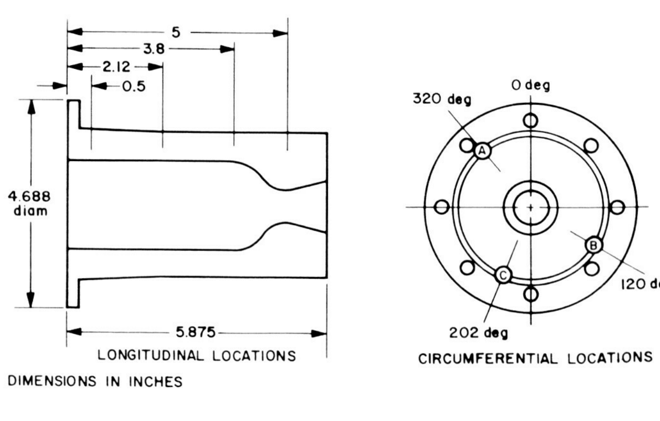

Fig. 4. Thermocouple locations on the ablative chambers

III. DESCRIPTIONS OF THE THRUST CHAMBERS

When received, each chamber was carefully inspected to verify compliance with the dimensional and weight requirements of the procurement specification. During this inspection, the throat diameter (and therefore the throat area) was recorded by tracing a ten-power-enlarged image of the throat circumference on a glass plate using a projection comparator. The distinguishing features of each chamber were then photographed. These inspection records and photographs plus the drawings, computations, and descriptions furnished by the suppliers, constituted the pretest file of data on the design and "as received" condition of the chamber.

In the following paragraphs, the chamber designs are reviewed in terms of their major design features. Table 1 is a summary of the principal details of interest, i.e., weight, overall dimensions, type of throat, and structural container used.

Table 1. Principal design characteristics of the five chambers

Source	Overall dimensions, length × diameter, in.	Weight, lb	Type of throat	Type of container
AGC	$5\frac{7}{8} \times 4\frac{11}{64}$	4	Silicon carbide	Glass roving
AVCO	$6\frac{1}{2} \times 5\frac{5}{8}$	$7\frac{1}{4}$	Silicon carbide	6061-T6 aluminum alloy
HITCO	$5\frac{7}{8} \times 3\frac{11}{32}$	2	Molded carbon cloth	Glass roving
MAP	$5\frac{7}{8} \times 3\frac{5}{16}$	$4\frac{1}{4}$	Molded carbon	Titanium alloy
TRW	$5\frac{15}{16} \times 4\frac{13}{16}$	8	Molybdenum	Stainless steel

A. *Aerojet-General Corporation (AGC) Chamber*

The Aerojet-General thrust-chamber assembly, shown in Fig. 5, used a throat insert of silicon carbide. The one-piece ablative chamber liner and exit cone were molded around the throat insert. The liner material was oriented high-silica roving tape impregnated with a rubber-modified phenyl-silane resin. A 30-deg (relative to chamber center line) tape orientation was used for the

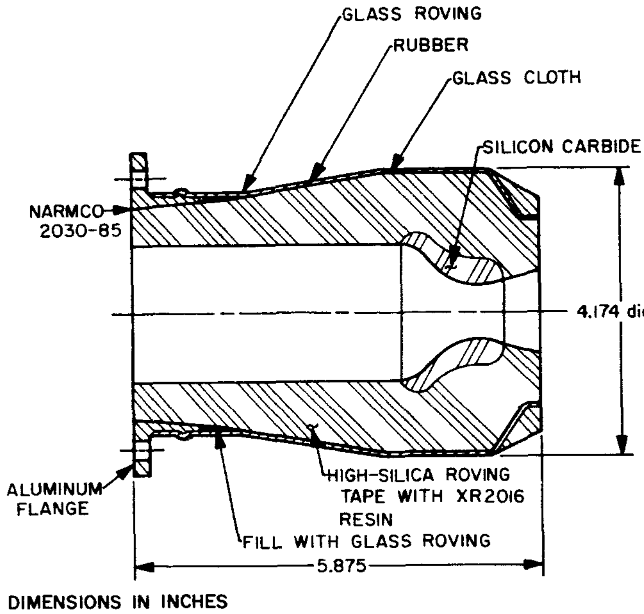

Fig. 5. Aerojet-General ablative-chamber design

ablative chamber from the injector end to the forward end of the throat insert. From this plane to the aft surface of the insert, the tape angle was gradually changed to 45 deg to the centerline, the tape angle used at the exit-cone section of the chamber. The entire assembly was covered with a laminated overwrap of longitudinally oriented glass fabric and epoxy resin. The aluminum mounting flange was sandwiched between the glass-fabric laminate and bonded to this glass fabric with a high-temperature filled epoxy resin. A gas-tight rubber bag (membrane) surrounded the laminated-glass-fabric overwrap (and chamber assembly) to prevent radial (outward) hot-gas leaks. Over the rubber covering, two layers of hoop-wound, resin-preimpregnated-glass roving were wrapped for structural support.

The technique and materials used to fabricate this chamber assembly apparently caused the formation of resin-rich and resin-starved areas. Upon curing, this improper distribution of the resin resulted in the formation of voids or pores. Figure 6 is a close-up view of a portion of the rear surface of the chamber where some of these pores intersect the external surface.

B. *AVCO Chamber*

The AVCO thrust-chamber assembly, shown in Fig. 7, used two ablative inserts, both with a 50-deg reinforcement orientation, made from a precharred silica cloth

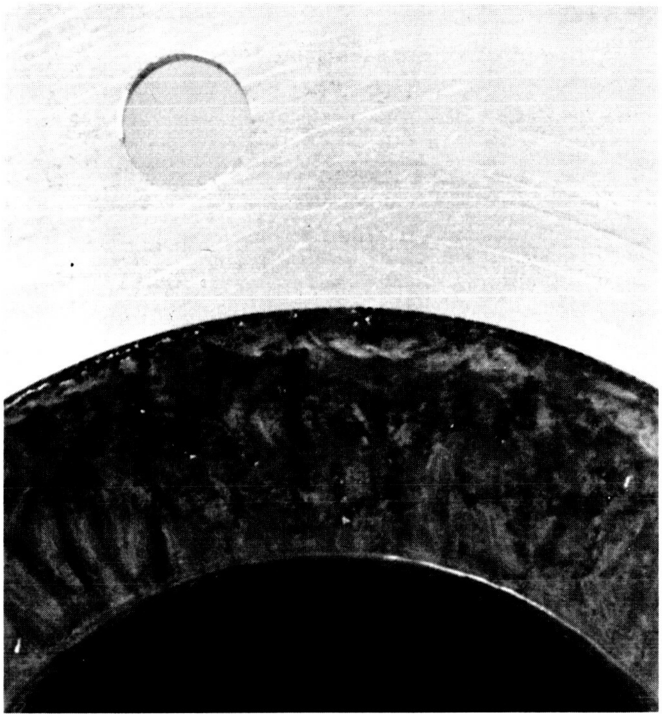

Fig. 6. Local pores and voids in the Aerojet-General ablative chamber (before firing test)

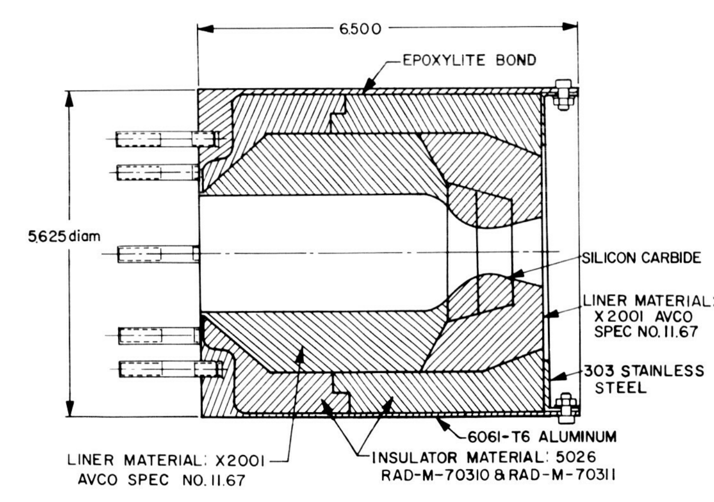

Fig. 7. AVCO ablative-chamber design

reinforced with a modified epoxy novolac resin system. The two-piece, silicon carbide throat insert was contained by the aft ablative insert. Both ablative inserts were surrounded by thermal insulators made from a low-density insulating material composed of silica fibers with phenolic microballoons in a modified epoxy novolac resin system. The ablative inserts and thermal insulators were bonded to each other and to the outer structural aluminum (6061) shell by means of an epoxy-type resin.

C. H. I. Thompson Fiber Glass Co. (HITCO) Chamber

The HITCO thrust-chamber assembly consisted of an ablative liner, backup insulation, laminated structural support, and aluminum mounting flange, in the configuration shown in Fig. 8. The molded ablative liner was made with plies of carbon cloth and phenolic resin, with the plies apparently oriented about 70 deg relative to the chamber center line. The throat and exit-cone sections were part of this molded ablative liner (i.e., no other pieces were used to form the throat). The backup insulator was flat-wrapped onto the ablative liner using a phenolic-resin-preimpregnated Refrasil material. The aluminum flange was bonded to both the ablative liner and thermal insulator, with a glass-cloth and phenolic resin laminate covering to supply structural support.

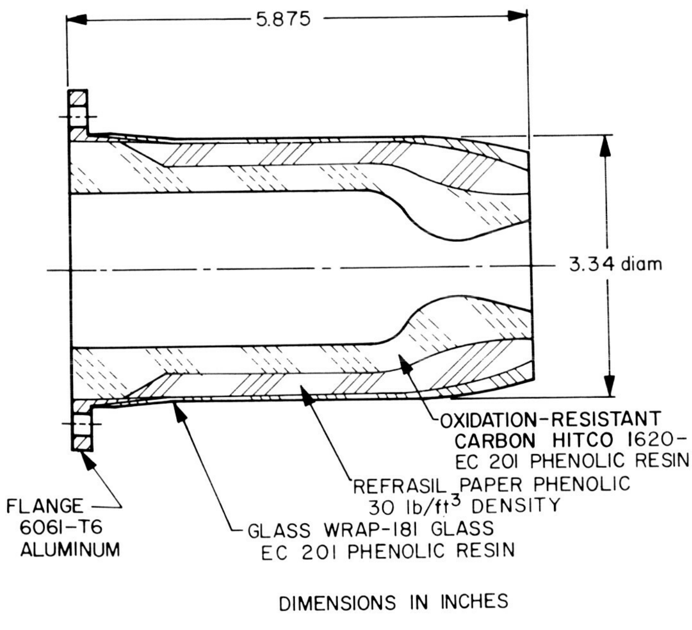

Fig. 8. H. I. Thompson Fiber Glass Co. ablative-chamber design

D. Magnesium Aerospace Products Company (MAP) Chamber

The MAP chamber design is illustrated in Fig. 9. It used a one-piece molded ablative chamber liner and throat. This molded piece was contained in a tapered molybdenum support sleeve. The molybdenum-sleeve and ablative-liner assembly was covered with a laminate

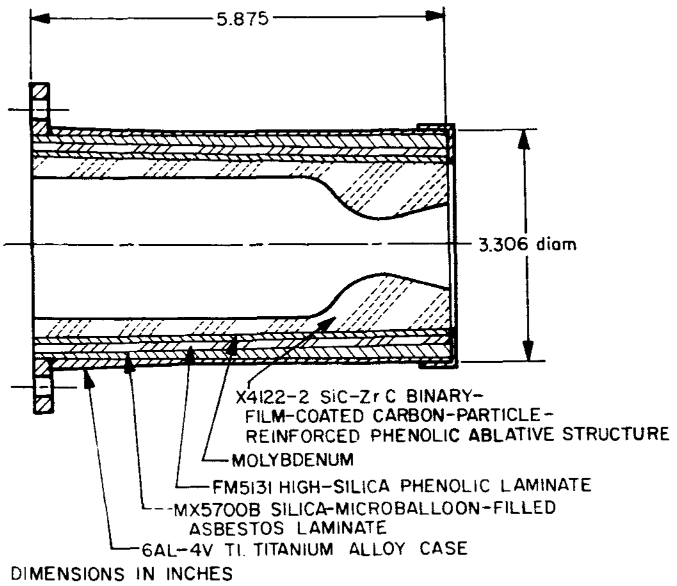

Fig. 9. Magnesium Aerospace Products ablative-chamber design

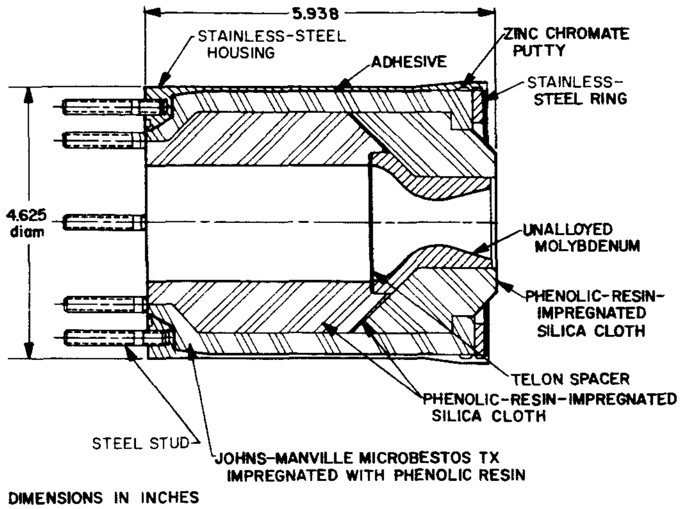

Fig. 10. Thompson Ramo Wooldridge ablative-chamber design

of high-silica fabric and phenolic resin for additional strength. A thermal insulator was used between the outer titanium alloy structural shell and the inner ablative-liner and sleeve assembly.

The ablative liner and throat piece were molded from a MAP-developed, SiC–ZrC binary-film-coated carbon-particle-reinforced phenolic resin. The thermal insulator was made of a silica-microballoon-filled asbestos paper and phenolic resin material.

E. Thompson Ramo Wooldridge, Inc. (TRW) Chamber

The TRW ablative-thrust-chamber design (Fig. 10) used a 45-deg-oriented chromic-oxide-coated silica cloth and phenolic resin material for both the ablative liner and throat-insert support liner. The ablative liners were overwrapped with a thermal insulation of TX Microbestos[6]. The machined plastic and molybdenum assembly was then bonded into a steel support shell. This shell had an injector-chamber sealing surface and mounting studs at one end, with a screwed-in retainer at the nozzle end.

[6]Johns-Manville Corp., New York, N. Y.

IV. RESULTS

All five tests were terminated because of an excessive decrease in chamber pressure due to nozzle-throat erosion or chamber failure. For four of the tests, this decrease was the result either of throat erosion or gas leakage around the throat insert. In one case, the chamber-injector mounting-flange attachment failed, permitting the chamber to separate from the injector. Although the test results are not highly encouraging, it should be realized that the test conditions were quite severe for ablative-type chambers since the chamber pressure was 150 psia and no film or fuel-rich "barrier" cooling was used.

Results of the test firings are summarized in Table 2 and discussed in the following paragraphs. Figure 11 should be consulted to understand the basic nomenclature used in describing the pertinent facts about the time and pressure relationships. This plot, which shows

Table 2. Summary of ablative-chamber test data

Source	Initial steady-state chamber pressure P_c, psia	Mixture ratio r, $(\dot{w}_{ox}/\dot{w}_f)$	Time to start of P_c decay,[a] T_1, sec	Time to P_c = 120 psia,[a] T_2, sec	Slope of P_c decay, psi/sec $\left(\frac{P_c - 120}{T_2 - T_1}\right)$	Maximum temperature during test, °F	Maximum temperature, postrun, °F
AGC	150	1.24	130	153	1.2	130° at 154 sec	475° at 450 sec
AVCO	152	1.20	263	590	0.1	405° at 590 sec	412° at 610 sec
HITCO	156	1.20	7	23[b]	2.2	67° at 23 sec	
MAP	164	1.22	8	47	1.1	254° at 57 sec	571° at 111 sec
TRW	150	1.18	160	515	0.1	240° at 510 sec	340° at 940 sec

[a]Refer to Fig. 11 for an explanation of these parameters.

[b]Attachment flange failed.

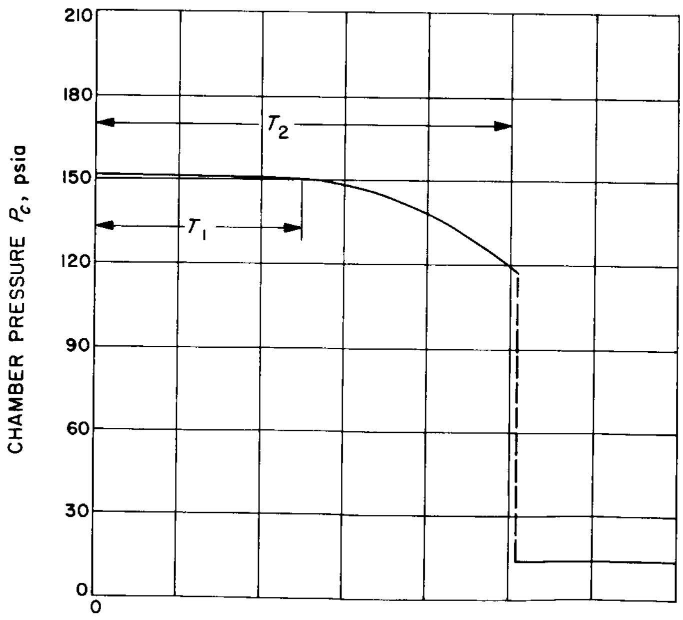

Fig. 11. Characteristic plot of pressure in an ablative chamber as a function of test duration

the characteristic shape of the chamber-pressure record, defines T_1 as the time at which pressure decay starts and T_2 as the time at which the chamber pressure has decreased to 120 psia.

After the chambers had been test fired, they were each weighed and photographed, and a tracing was made of a ten-power-enlarged image of the throat area as projected on the comparator. From this tracing, the final throat area was measured with a planimeter. In addition, the extent of local erosion could be determined by comparing the final throat outline (tracing) with the prerun (circular) outline. Each chamber was then sectioned, photographed, and examined for internal condition (i.e., measurement of char depths, and erosion patterns).

A. AGC Chamber Test

As shown in Fig. 12, chamber pressure in the AGC chamber stabilized at about 150 psia within 20 sec, followed by a decrease of only 2 psi during the next 110 sec of testing. At 132 sec the chamber pressure began to decrease rapidly until the test was terminated at 153 sec when it was believed that the chamber throat had failed. Postrun inspection of the chamber showed that the throat insert was in reasonably good shape despite the fact that a portion in the forward side was broken off. At least three good-sized holes were found around the outside of the throat insert and these were apparently the main cause of chamber-pressure loss. Figure 13 shows the change in the nozzle throat as a result of the firing.

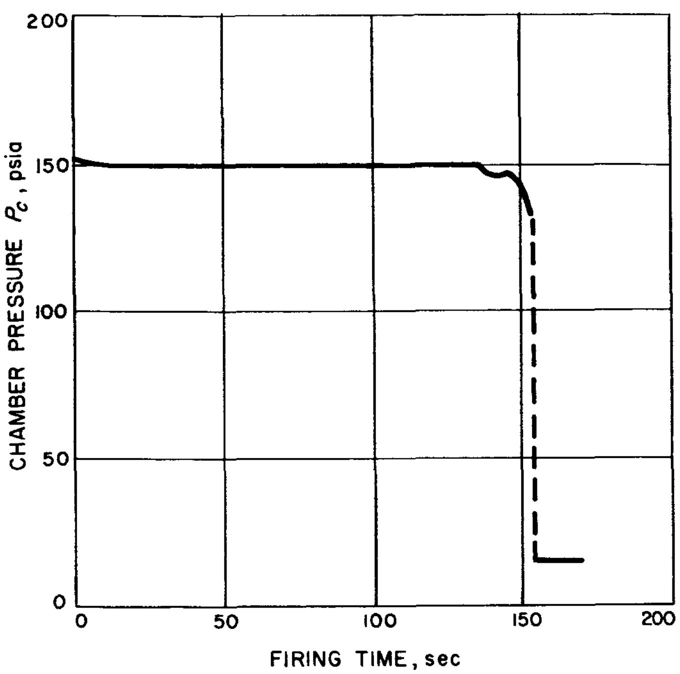

Fig. 12. Record of chamber pressure during the firing test of the Aerojet-General ablative chamber

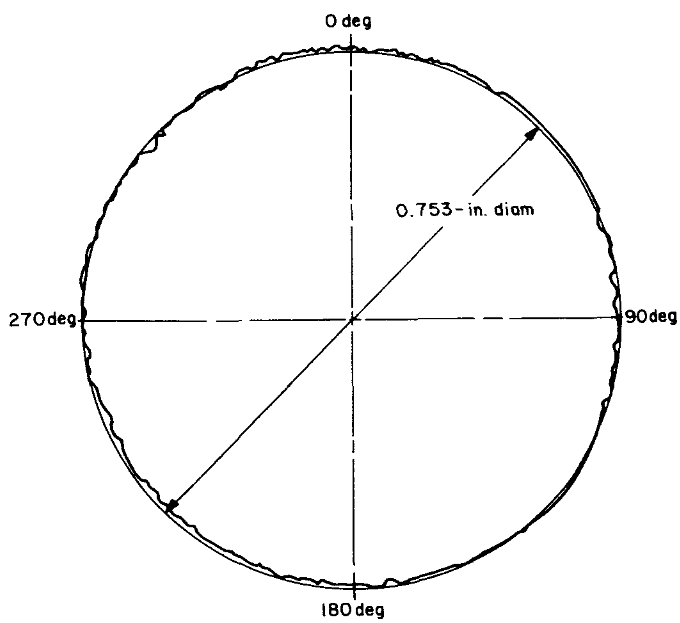

Fig. 13. Nozzle-throat area in Aerojet-General ablative chamber, before and after the firing test

The axial thrust changed very little during the run; its variation was similar to the chamber pressure, i.e., essentially constant with a rapid decrease just before the end of the test.

During the first 130 sec of the test, one of the side-thrust components showed some changes[7] that cannot be easily explained since chamber pressure and axial thrust were constant during this time. After 132 sec the thrust changed direction and the rapid shift was probably related to the cause of the rapid decay of chamber pressure, i.e., the burning through of several holes around the throat insert.

There was a total weight loss of about 0.4 lb during the 153 sec of firing time. This is a 9% weight loss. Although the ablative liner was badly eroded in several places around the throat insert (Fig. 14), it had only two significant longitudinal grooves in the combustion chamber area. One of these grooves was quite deep and the underlying material was charred through to the over-wrap. Either during or following the run, sufficient heat penetrated the wall to cause cracking and separation of the outer structural glass wrap. Despite this localized hot spot, shown in Fig. 15, the rubber gas-tight bag apparently had not failed up to the time of shutdown. It is almost certain the thrust chamber would have failed structurally because of charring through the outside structural wrap within another 2 min of firing. After the chamber was sectioned, the char depth was found to be fairly uniform along the entire length of the chamber with the exception of the deeply eroded groove area.

The maximum external-wall temperature measured during the run was 142°F. This temperature was detected by the thermocouple closest to the injector mounting flange. The maximum heat-soak temperature (475°F) was measured about 300 sec after the end of the test at a location about midway down the cylindrical length of the combustion chamber.

B. AVCO Chamber Test

The AVCO chamber ran for about 4½ min with little or no change in chamber pressure (Fig. 16). The chamber pressure stabilized at 152 psia after about 50 sec and remained constant for 263 sec followed by a decline to 117 psia at the termination of the test at 600 sec.

Inspection of this thrust chamber following the test revealed two deep, locally eroded portions of the throat insert about 180 deg apart, with little or no evidence

[7]Because of a slight amount of cross-coupling between axial- and side-thrust loads in the six-component measuring system, only changes in side thrust are mentioned in this and the descriptions that follow.

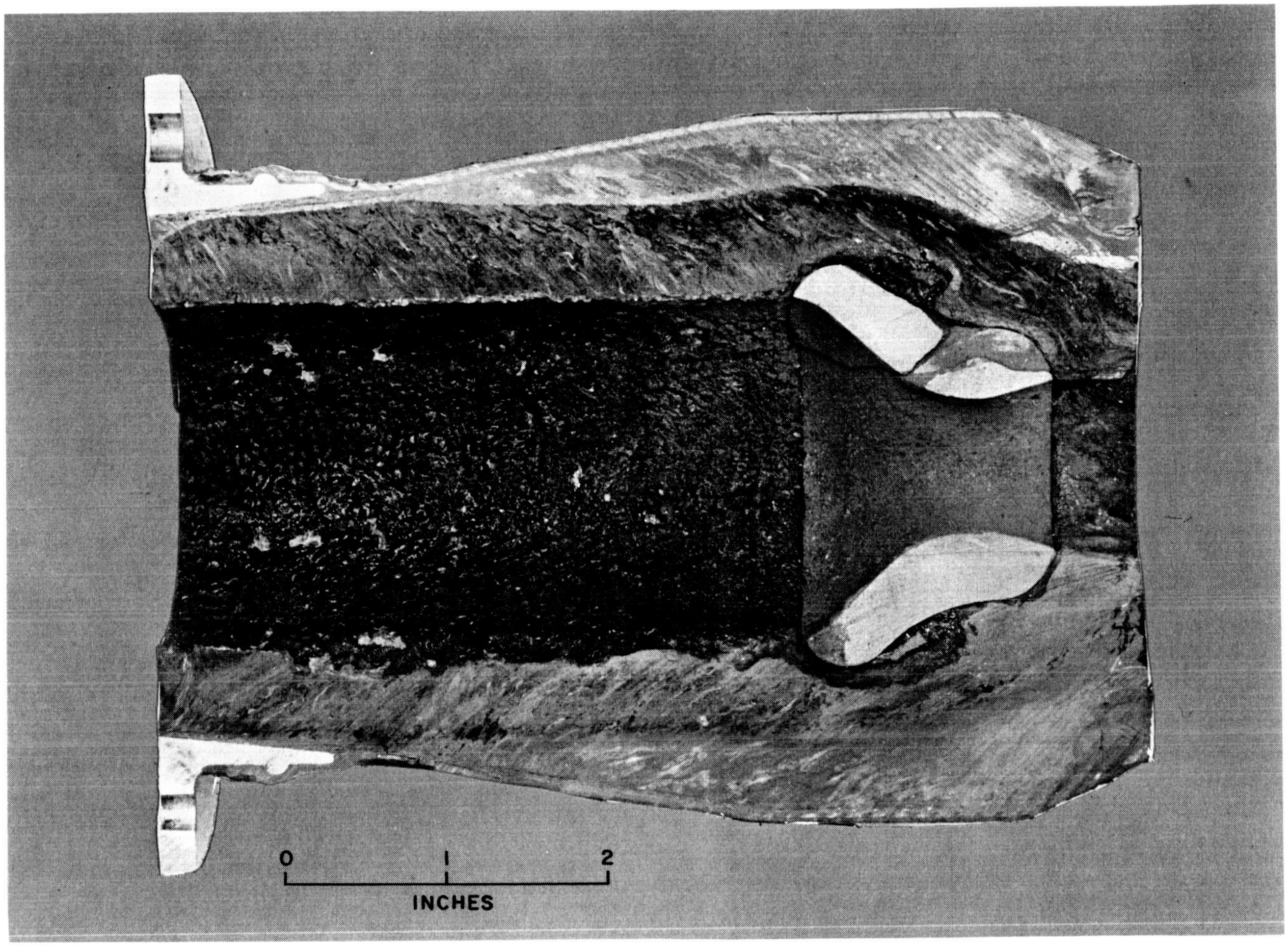

Fig. 14. Cross section of Aerojet-General ablative chamber after the firing test, showing the cracked nozzle-throat insert and the char depth

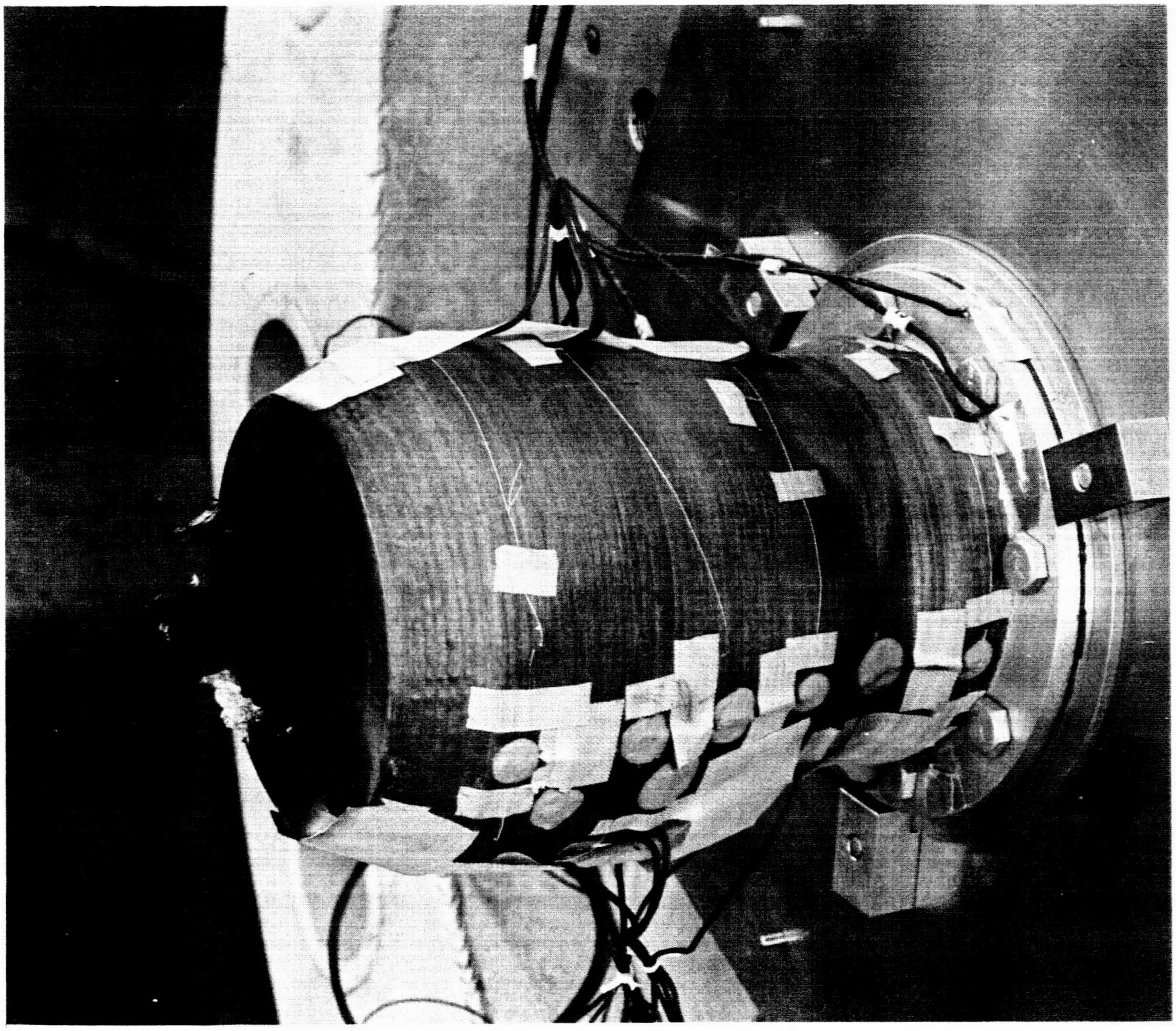

Fig. 15. External view of Aerojet-General ablative chamber after the firing test, showing the local hot spot and wall separation

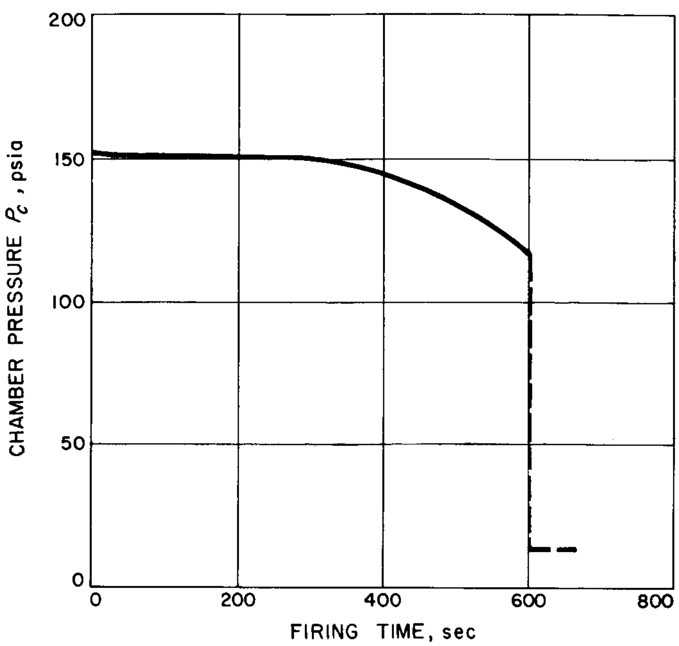

Fig. 16. Record of chamber pressure during the firing test of the AVCO ablative chamber

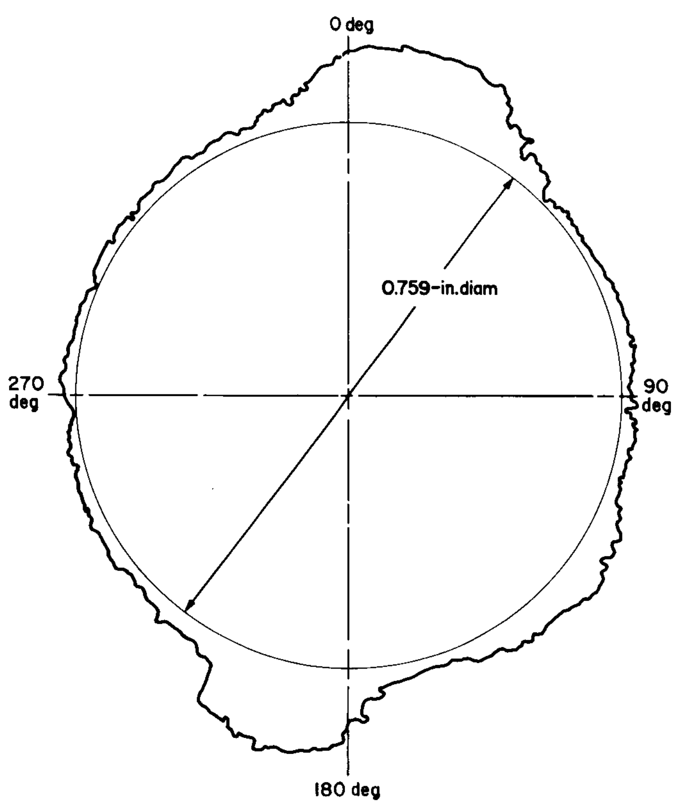

Fig. 17. Nozzle-throat area in AVCO ablative chamber, before and after the firing test

of hot-gas leaks around the throat-insert assembly (washers). Figure 17 shows the nozzle-throat outline before and after the test.

After the chamber was sectioned, the throat inserts were found to be cracked in several places, with one badly eroded area between the two insert washers. Figure 18 is a view of a section of the chamber.

There were five particularly deep longitudinal channels eroded in the inside surface of the chamber. The oblique view of the injector end of the chamber seen in Fig. 19 shows two of these channels. The appearance of these channels suggests a "washing out" of the ablative material by the local combustion or propellant-spray conditions.

The record of the axial thrust measured during the test resembles that of the chamber pressure. However, it shows a decline of thrust beginning about 110 sec before the start of the decline of the chamber pressure.

The side-thrust record shows a change beginning about the same time as the axial thrust, with a continuous shift up to about 100 sec before shutdown, when the side thrust changed direction and shifted continuously until shutdown.

This chamber assembly had a total weight loss of only 10%, although the post-test appearance of the chamber, particularly after it was sectioned, would have prompted a guess of a greater weight loss.

The highest outside-wall temperatures, both during and after the test, were measured at the thermocouple locations closest to the injector flange. In addition, there was a 250° longitudinal temperature gradient from the injector to the exit end of the chamber shell. A possible reason for this gradient was the design of the injector mounting flange and the lack of thermal insulation between the ablative chamber liner and the combustion zone. The rate of temperature rise at the injector end of the chamber was fairly severe, and if the chamber pressure had not dropped, this test would have been terminated because of excessive external-wall temperature (much greater than 400°F).

Inspection of the sectioned chamber assembly (Fig. 18) reveals severe cracks and voids in the thermal insulator and ablative chamber liner. An appreciable rearward shift of the throat-insert and ablative-wall assembly caused a very large crack in the ablative material just forward of the throat insert.

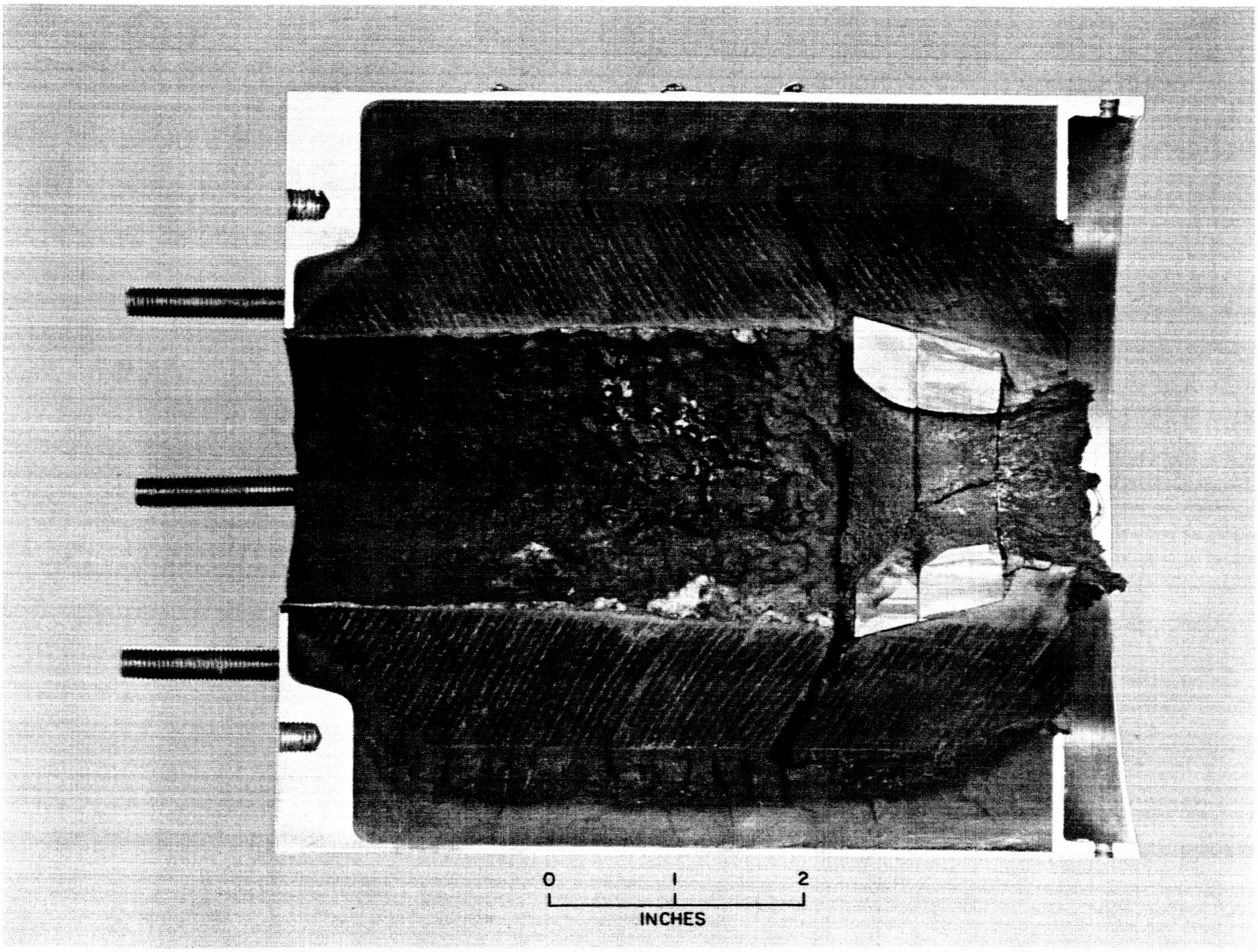

Fig. 18. Cross section of AVCO ablative chamber after firing test, showing the cracked nozzle-throat insert and char depth

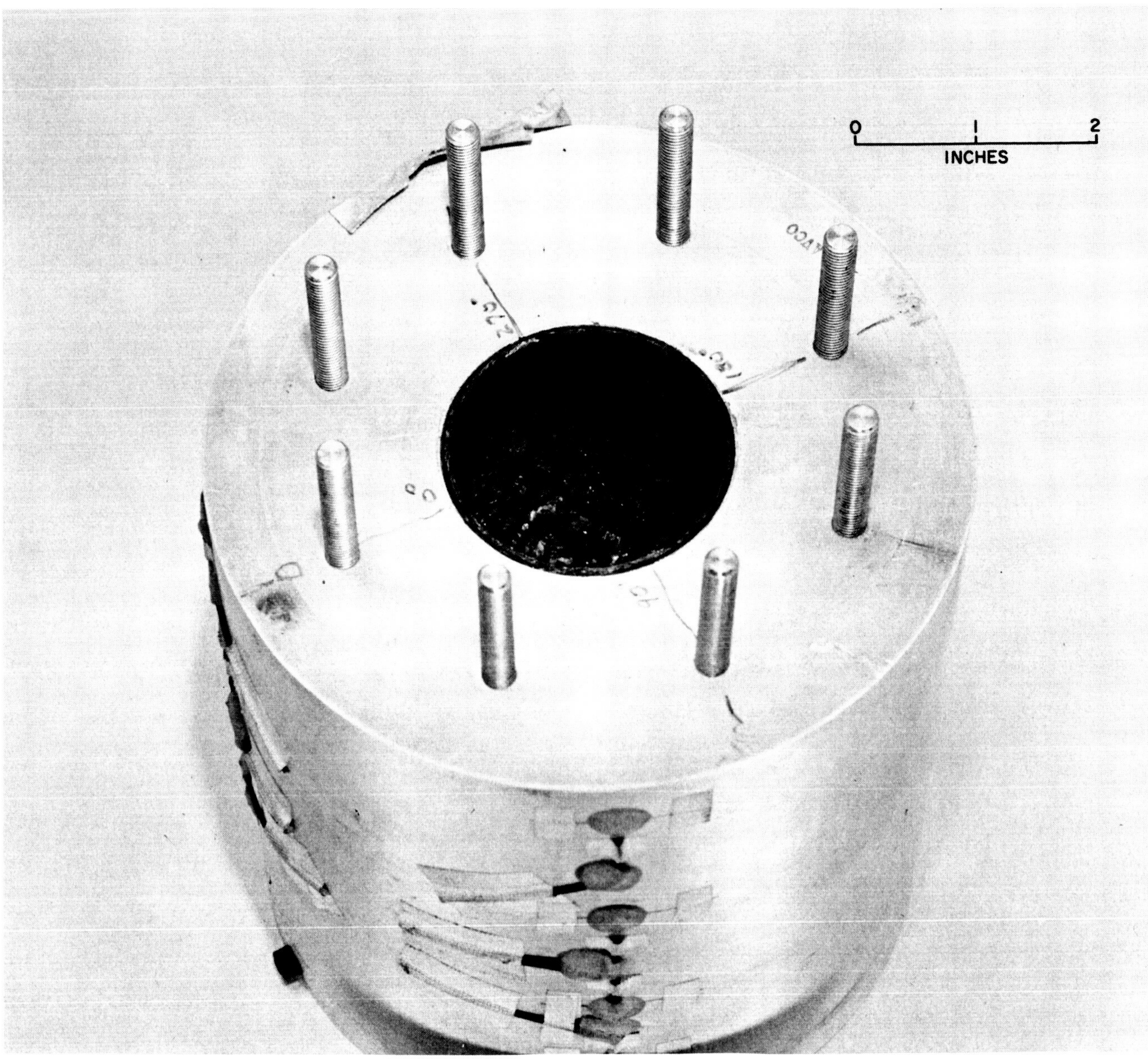

Fig. 19. Oblique view into AVCO ablative chamber, showing a localized region of intense erosion

The ablative liner had a very liquid-appearing surface (i.e., an appearance similar to water on an oiled surface), with considerable material washed down toward the insert assembly. There was also a deposit buildup on the surface of the insert downstream of washed-out areas of the chamber.

The thermal-insulating material surrounding the ablative liner was charred about half-way through. As can be noted from the picture of the sectioned chamber half (Fig. 18), the char was about the same depth over most of the length of the chamber.

C. *HITCO Chamber Test*

The test of the HITCO ablative thrust chamber was abruptly terminated after 23 sec when the chamber separated from the aluminum mounting flange. The exact cause for this structural failure is not known, but the outer structural cloth wrap was apparently improperly bonded to the aluminum flange, with the thrust chamber held together only by the inner ablative liner and thermal insulator. Failure probably occurred when the charring and heat had weakened the ablative-liner and insulator assembly to the point where the pressure load could not be sustained.

The chamber pressure was stabilized for only a few seconds at 156 psia before it started to decrease to a value of 128 psia at the time the thrust chamber failed. The chamber-pressure decay was nearly straight-line and up to the time of chamber failure, it had declined over 18% for the 23 sec of test time. Figure 20 is a plot of the chamber pressure vs time.

Comparison of the throat area before and after the firing test (Fig. 21) showed a total throat-area increase of 22% or nearly a 1% increase per second of test time. Extrapolation of the chamber-pressure decay indicated that without the structural failure of the chamber, the test would probably have been terminated after about 30 sec total duration when the pressure would have reached 120 psia.

Despite the severe throat erosion, the indicated axial-thrust decrease was only 2 lb; this is only 2.4% of the nominal value of 84 lb. The near-constant thrust is a result of the compensating effects of pressure decrease and area increase, coupled with a slightly increased propellant flow. The change of axial thrust was essentially proportional to the change in chamber pressure; i.e., a short period of constant thrust followed by straight-line decrease.

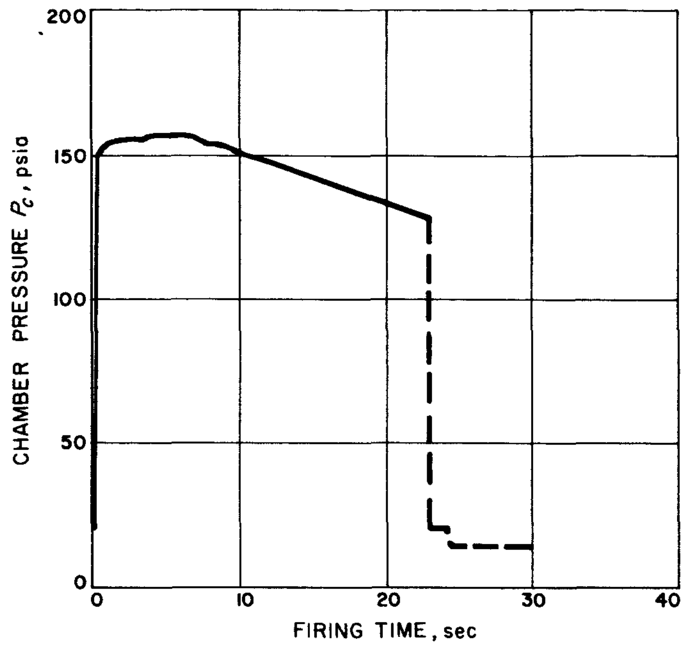

Fig. 20. Record of chamber pressure during the firing test of the HITCO ablative chamber

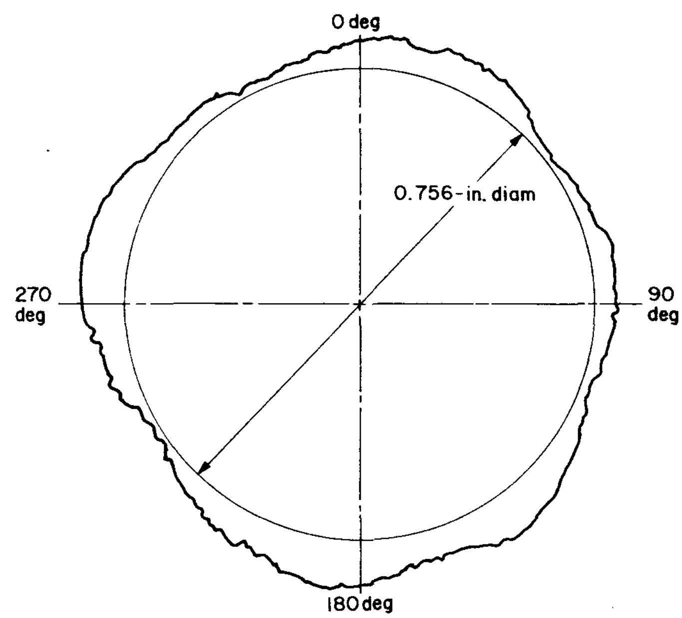

Fig. 21. Nozzle-throat area in HITCO ablative chamber, before and after the firing test

The thrust-chamber-assembly weight loss was nearly 2% during the 23-sec firing test. Compared to the two most successful chambers tested, the HITCO chamber had five times as much weight loss for the short test duration. Erosion of the ablative chamber was fairly uniform with the only local highly eroded areas found in the throat.

The external-wall temperatures changed little, if any, during the test. The three thermocouples located on top of the aluminum flange–fiberglass overwrap did not indicate the occurrence of an excessive temperature, which would have caused the failure of the bonded joint.

The sectioned chamber (Fig. 22) showed some cracking and separation of the ablative-liner material near the throat and exit portions of the chamber. This damage may have been caused when the chamber struck the concrete deck after it separated from the aluminum mounting flange (it first landed on one corner of the exit end of the chamber).

The design of this thrust chamber had two major weaknesses: an inadequate method of attaching (bonding) the outer structural glass fabric to the metal flange, and a lack of thermal insulation between the ablative chamber liner and the flange.

The extremely smooth metal flange provided a poor bonding surface, and only a thin layer of silica tape was wrapped over the flange. The tape was applied in a rosette pattern, with strips of material running longitudinally along the chamber wall. This is a hand lay-up technique, which does not permit sufficient tension to be applied to the tape for a tight fit.

The lack of thermal insulation between the chamber liner and the metal flange would have led to excessive heating of the flange and structural overwrap. Furthermore, the molded thermal insulator between the outer structural glass fabric and the inner ablative liner showed voids and irregular shape and density (Fig. 22). What effect these deficiencies would have had on the ultimate operation of the chamber is not known since the test duration was so short.

D. MAP Chamber Test

The molded ablative chamber liner and throat used by MAP in their chamber design eroded quite severely.

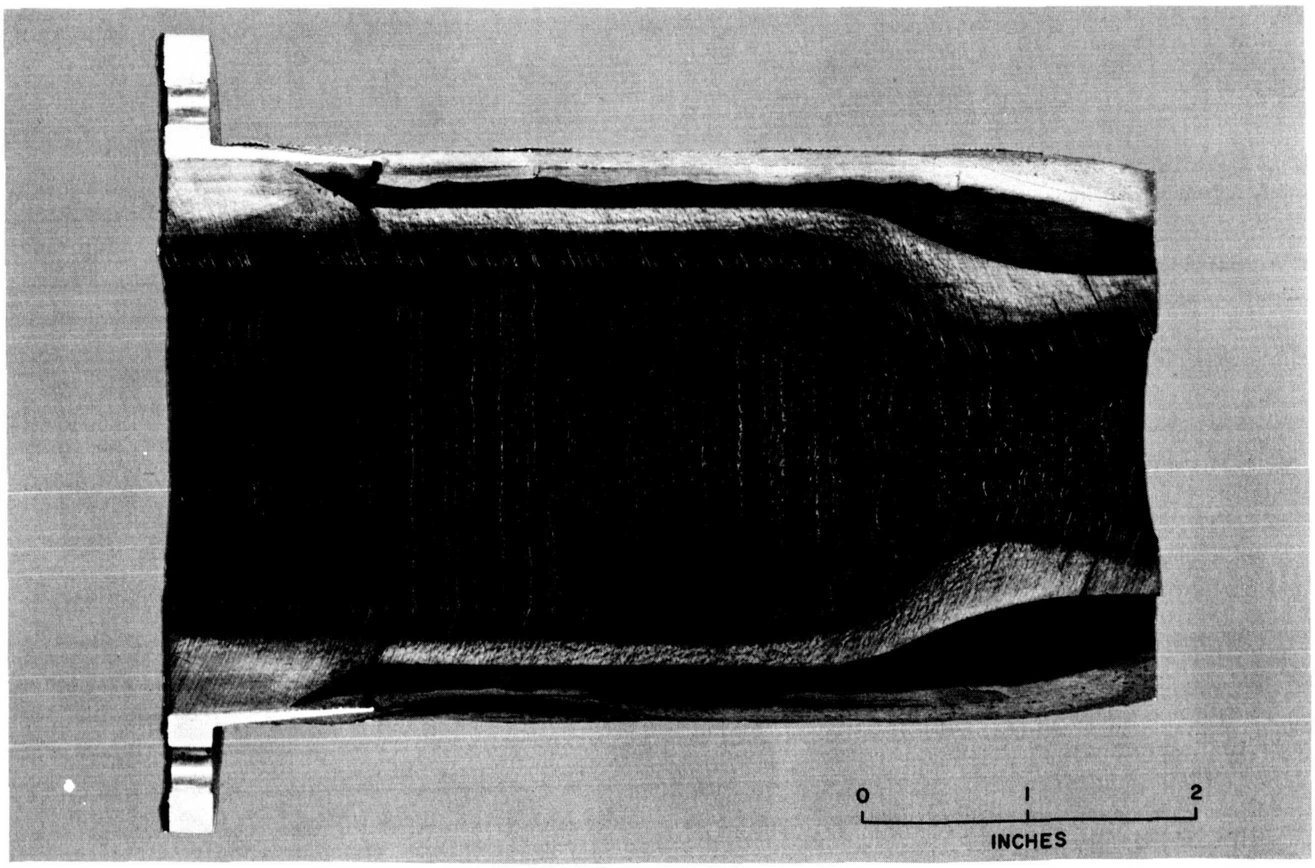

Fig. 22. Cross section of HITCO ablative chamber, showing the area of separation

The throat apparently had an initial shrinkage followed by a fairly rapid enlargement. The chamber pressure record (Fig. 23) shows an initial pressure rise to a value of 164 psia (nearly 10% higher than anticipated), followed by a continuous decrease to 113 psia at the end of the test.

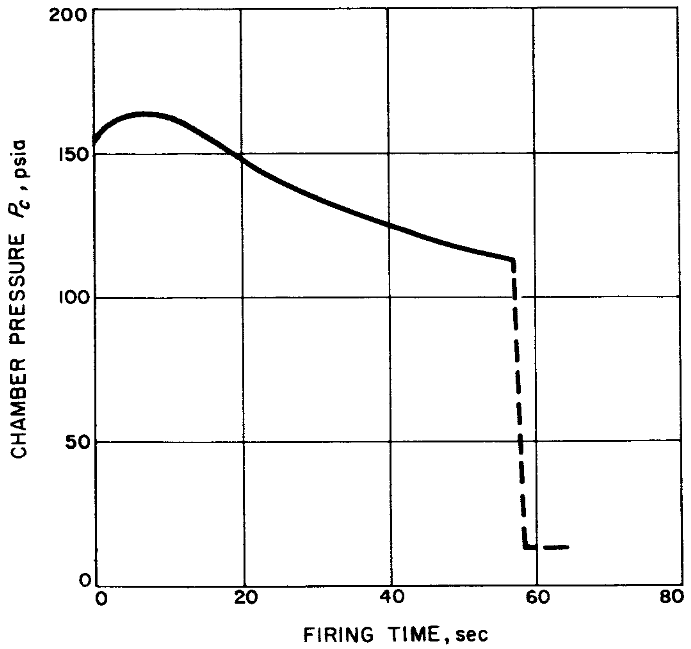

Fig. 23. Record of chamber pressure during the firing test of the MAP ablative chamber

During the 57-sec test, there was a total throat-area increase of 41%. Total weight loss was over 6%, which when compared on a test-duration basis, is the highest weight-loss rate of the five chambers tested.

The erosion pattern of the throat, shown in Fig. 24, was fairly uniform with only two areas slightly more eroded than the average. The erosion of the chamber section was not so uniform. The sectional view in Fig. 25 shows that the majority of the material loss occurred starting approximately 1½ in. downstream of the injector with "islands" of material separating rather large and deep channels where much material was lost.

The axial- and side-thrust records appeared to follow the same kinds of variations as the chamber pressure. Both leveled off for a short time and then changed continuously as the throat eroded and the chamber pressure steadily decreased until the end of the test. The axial thrust shows nearly a 2% change, while the side-thrust change was over 3½ times the initial offset value.

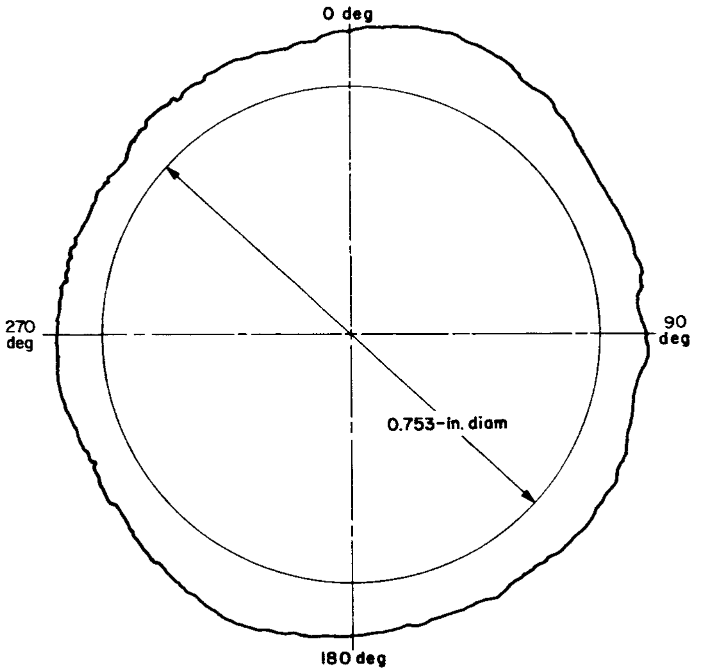

Fig. 24. Nozzle-throat area in MAP ablative chamber, before and after the firing test

The highest outer-wall temperatures were measured with the thermocouples closest to the injector. Their rate of temperature rise was rapid. At the end of the test (after 57 sec), the temperature had already reached 250°F and the maximum post-test temperature of 540°F (same location) was reached only 40 sec later. Along the length of the chamber, the outer-wall temperature decreased toward the exit end of the chamber. At the time of test termination, the temperature at the injector end was 2½ times the surface temperature at the aft end.

The sectioned chamber (Fig. 25) also revealed local cracking and separation of the ablative liner. The liner had two large cracks (traverse to the axis of the chamber) on each side of the throat and there were many small cracks, particularly along the surface of the molybdenum retainer sleeve.

This chamber design is judged to be unsatisfactory for several reasons; two in particular are: (1) the choice of material for the ablative liner, which was the same for both the chamber and the throat, was poor, as evidenced by severe erosion in both areas, and (2) there seems to be an insufficient amount of thermal insulation between the ablative liner assembly and the outer structural shell to give effective temperature control.

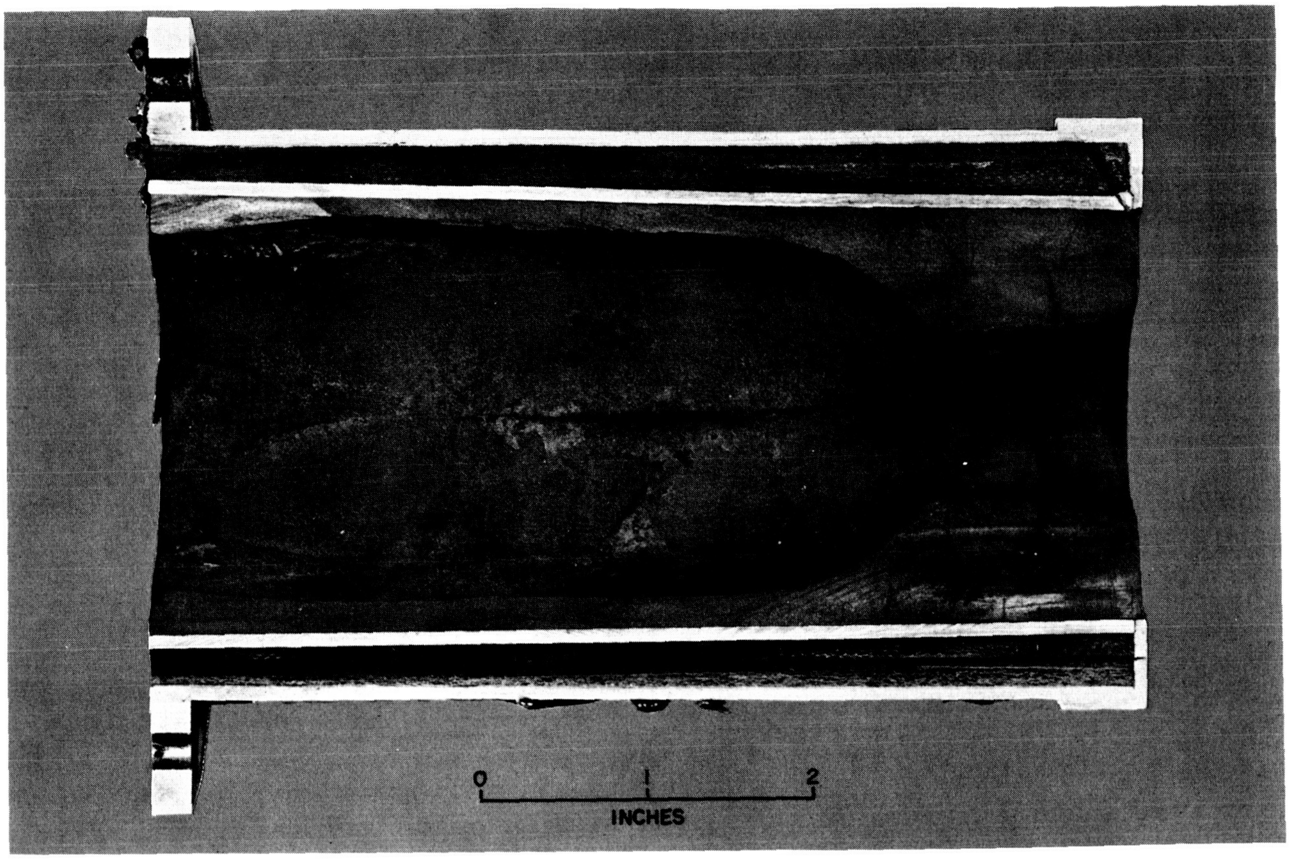

Fig. 25. Cross section of MAP ablative chamber after the firing test

E. TRW Chamber Test

Failure (burn through) of the throat insert, with the associated loss of chamber pressure, caused the termination of the test of the TRW ablative thrust chamber. As can be seen in Fig. 26, the chamber pressure stabilized at 150 psia and remained essentially unchanged for 160 sec. At this time, erosion of the throat insert began and the chamber pressure continually decreased until it reached 124 psia after 508 sec, when the throat insert completely failed (burned through). After this, the chamber pressure dropped to 112 psia at 520 sec, at which time the firing was stopped. The throat area was found to have increased a little over 19% during the test; Fig. 27 is a diagram of the nozzle-throat area before and after the firing.

The axial thrust remained essentially constant for the first 270 sec of the test despite the loss of chamber pressure, which started around 160 sec. Until 270 sec had elapsed, the indicated thrust decrease was only 1½%. By the end of the test, axial thrust had decreased 3.6% from the initial value.

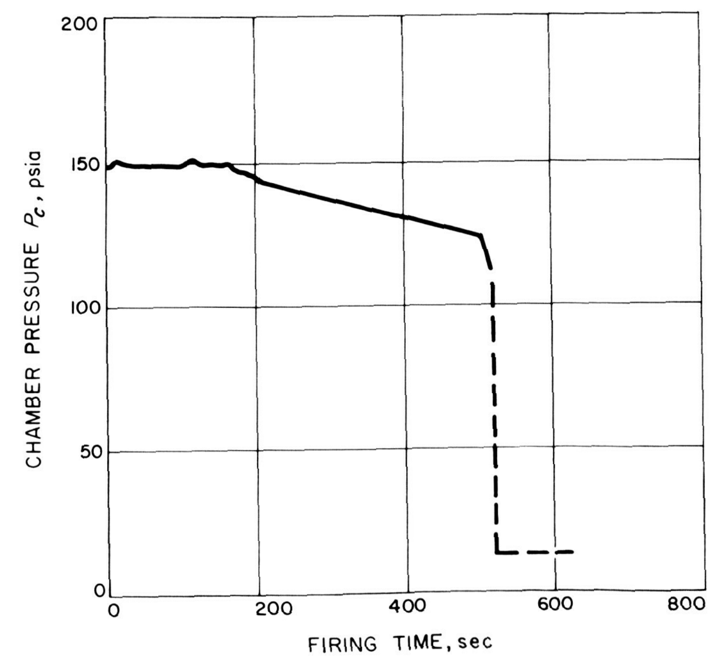

Fig. 26. Record of chamber pressure during the firing test of the TRW ablative chamber

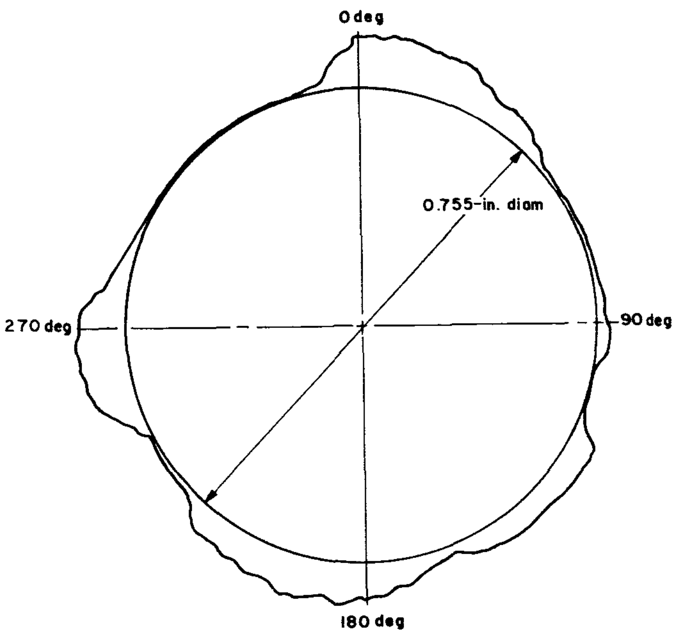

Fig. 27. Nozzle-throat area in TRW ablative chamber, before and after the firing test

An unexplained 3-psi rise in chamber pressure occurred at about 115 sec. Signs of this disturbance were also found in the thrust records and it appeared to take about 20 sec for the pressure to return to the prior level.

There was no completely stabilized value of side thrust during the early part of the test. Between 160 and 260 sec, the side thrust was fairly constant, but following this period there was a continuous shift until a peak was reached at about 380 sec; then the shift in side thrust reversed direction. At about 440 sec this shift of side thrust again reversed direction and shifted continuously for 60 sec, ending when an abrupt and rapid change of direction of side thrust occurred just before the test termination.

Of the five chambers tested, the TRW chamber showed the smallest weight loss per second of firing time. Total weight loss was about 7.4% over the 520-sec test duration.

The appearance of the sectioned chamber can be seen in Fig. 28. There are no cracks or voids visible in the ablative portion of the sectioned chamber. The inside wall surface had a "bubbly" appearance with some areas showing more loss of material just forward of the throat insert. There was a heat or char discoloration through the entire thickness of the oriented ablative material, but not through the thermal-insulating material. Char thickness is fairly uniform along most of the length of the chamber.

The throat insert was badly eroded in four places and in at least one place it was completely burned through. The surface of the convergent portion was badly pitted, with one area on the forward (injector) side showing signs of a possible attack by the ablative chamber residue. There was no evidence of cracking of the throat insert. There was little or no buildup of deposits on the throat-insert surface. The nozzle-exit end of the throat insert was left in reasonably good condition, with little or no channeling or local erosion evident.

The injector end of the chamber heated up very rapidly and reached the highest temperature recorded during the test (240°F). One reason for this might be that the mounting flange provided little room for thermal insulation between the metal structural shell and the inside surface contacting the hot gas so the radial heat flow at the injector end of the chamber was quite substantial. Longitudinal temperature variation was about 100°F. A maximum postfiring temperature of 334°F occurred about 420 sec after the end of the test, at a point about midway down the chamber.

In general, the TRW chamber appeared to be a good design; its two weak points were the throat insert and the radial heat path of the metal mounting flange. (The latter shortcoming was almost intrinsic in the specification requirements calling out the flange dimensions and all of the chamber designs were weak to a degree from this fault.) The appearance of this chamber after firing, with the exception of the throat insert, was quite good and the ablative liner was probably in the best condition of any of the chambers tested.

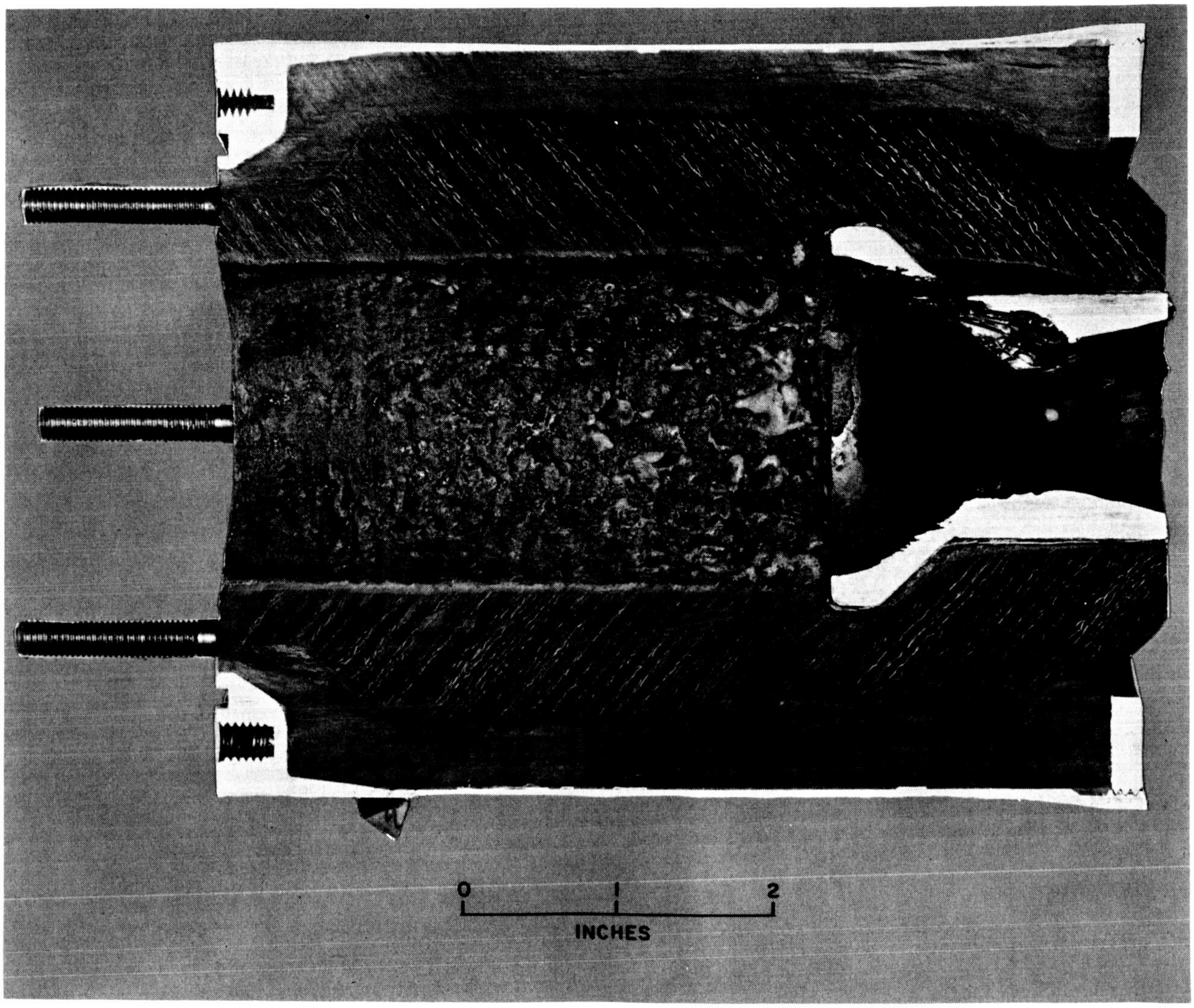

Fig. 28. Cross section of TRW ablative chamber after firing test, showing the undermining of the nozzle insert

V. CONCLUSIONS

None of the five chamber designs demonstrated a capability of very long burning times under the rather severe test conditions used in this program. These disappointing results may be attributed in part to the fact that the JPL procurement specifications (the written specification and the configuration control drawing given in the Appendix) used for this program were unfortunately lacking in some important details and ambiguous in others. One of the more serious omissions was the requirement that each chamber successfully pass a pressure-proof test as partial verification of a satisfactory design. The procurement specification also was not clear about the method of attachment and maximum allowable thrust-chamber outside diameter. Two vendors provided a large outside diameter and mounting studs, but the metal sealing surface used in these designs acted as a severe radial heat path that short-circuited the thermal insulation placed between the ablative inner liner and the outer metal shells. The designs of the three other chamber suppliers avoided the metal radial heat path, but used very small chamber outside diameters to provide mounting bolt clearance. With these smaller diameters, it was impossible to provide adequate thickness of thermal insulation between the inner (hot) ablative liners and the outside structures. All of the chamber designs experienced their highest outside-wall temperatures in the area adjacent to the injector mounting flange.

Two chamber designs (AVCO and TRW) gave nearly 10 min of service before catastrophic failure of their throat inserts. The silicon-carbide-type insert was more resistant to erosion than the coated-molybdenum design, giving 100 sec more burning time before chamber-pressure decay began.

None of the five chambers tested were in satisfactory condition for restart after their long-duration tests. The major reasons for this were (1) failure (burn through) of the throat inserts, (2) hot-gas leaks around the insert, (3) excessive throat-area increase, or (4) structural failure (chamber broke in two). Furthermore, the chamber liners were generally in poor condition for additional service because of material and structural integrity loss (due to erosion, cracks, and voids).

Visual examination of the chambers both before and after testing gave evidence of considerable variability of quality control in their manufacture. The chambers that appeared to be the most carefully designed and built (and were also the most costly) proved to be the most durable.

Ablative chamber liners constructed of carbon materials were not found to be suitable for use with the test propellant system and operating conditions. Despite the differences in design of the two chambers that used carbon-type liners and throats (HITCO and MAP), both suffered substantial throat erosion and loss of chamber-liner material in their very short run durations.

APPENDIX

Specification for Test Samples of One-Hundred-Pound-Thrust Ablative Chambers

1. Purpose

This specification details requirements which are to be satisfied by test samples of ablative thrust chambers. These samples will be used in a partial assessment of the current state of the art of designing and fabricating chambers in the size range needed for both manned and unmanned spacecraft. Conformance with this specification is necessary to assure that all chambers obtained can be test fired under the same conditions so that a valid correlation of the data can be made.

2. Configuration

The chambers shall conform to the geometry indicated on Fig. A-1. All unspecified dimensions are left to the discretion of the supplier.

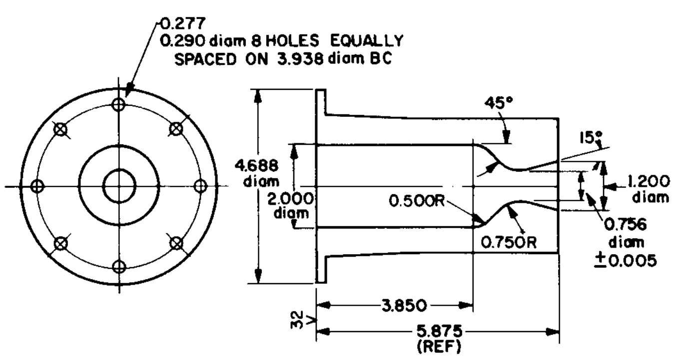

Fig. A-1. Control drawing of 100-lbf-thrust chambers

3. Construction

The supplier shall be responsible for determining the materials and fabrication procedures to be used in making the chamber. Hard materials may be used for liners or throat inserts if the supplier wishes to do so.

4. Design

The supplier shall be responsible for generating the design. The chamber should be strong enough to withstand a proof pressure of 225 psig prior to firing without any damage to the chamber or distortion of the injector mounting flange where the "o"-ring seal will be located. Sufficient space about the bolt holes must be left clear for easy access to the mounting bolts. The external surface should be reasonably smooth so that thermocouples may be attached. The entire chamber (nozzle, chamber, and mounting flange) shall weigh no more than eight pounds.

5. Cooling

The chamber must be ablative or heat-sink cooled. No propellant or fluid coolant will be used for regenerative, film, or transpiration cooling the chamber. Radiation and air cooling will be limited only by the external temperatures allowed.

6. Space Suitability

Although the evaluation tests will be conducted at ambient conditions, the supplier should recognize that the suitability of the chamber for repeated storage and firing cycles in space will be estimated. Engineering judgment concerning this matter will be considered in the overall evaluation of the chamber.

7. Test Conditions

The evaluation firing will be conducted at JPL's Edwards Test Station where the average barometric pressure is 13.7 psia and the ambient temperature ranges from +20 to +115°F. The chamber will be fired with an injector designed specifically for reproducible operation. Approximately one-third of a pound per second total weight of nitrogen tetroxide (N_2O_4) and hydrazine (N_2H_4) will be burned at a weight mixture ratio (O/F) of 1.2. A characteristic velocity of 5450 to 5550 feet per second will produce a (plenum) chamber pressure of approximately 150 psia. The test firing will be made in the horizontal position. The firing will continue uninterrupted until the chamber burns through or a prescribed external temperature limit is exceeded or a prescribed side thrust limit is exceeded. The allowable external temperature limit at any location will be 400°F. The allowable side thrust (90° from axial) limit will be five per cent of the axial thrust. The data used for evaluation will include side thrust versus time, axial thrust versus time, and external temperature versus time.

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