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Cyclic Hot Firing of Tungsten-Wire-Reinforced, Copper-Lined Thrust Chambers

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Cyclic Hot Firing of Tungsten-Wire-Reinforced, Copper-Lined Thrust Chambers

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Summary

This report describes an advanced thrust chamber liner material, a tungsten-wire-reinforced copper composite, for potential use in long-life, reusable rocket engines. By strengthening the chamber in the hoop direction, this material may prevent longitudinal cracking due to low-cycle fatigue, which has been observed in conventional homogeneous copper chambers, without reducing the high thermal conductivity that is essential when operating with high heat fluxes. The tungsten-wire-reinforced copper composite was used in two hydrogen-oxygen rocket thrust chambers to evaluate its performance as a reusable liner material. Test results show that both chambers failed prematurely by cracking but that the crack sites were perpendicular to the normal direction of cracking. These findings indicate a degree of success in containing the tremendous thermal strain associated with high-temperature rocket engines. In all cases, the composite liner failures were associated with drilled instrumentation ports, and no other damage or deformation was found.

Introduction

This work is part of NASA's continuing effort to extend the life of high-temperature, high-pressure rocket thrust chambers. At present, these chambers are fabricated from a high-thermal-conductivity material such as copper and are regeneratively cooled in order to maintain the chamber wall temperature at a reasonable level in this severe thermal environment. During a rocket engine firing, the thrust chamber liner is exposed to extreme heat fluxes (refs. 1 and 2). This environment causes the temperature of the hot-gas-side wall to rise rapidly. The temperature rise in this area, coupled with the relatively cool and fixed outer structure, causes the hot-gas-side wall material to experience very high thermal strains. These high thermal strains ultimately determine the reusable life of the thrust chamber liner.

In the past, these high thermal strains would accumulate in the thin wall between the coolant side and the hot gas side and cause a necking down of material and a longitudinal crack formation after a relatively low number of cycles. To extend the life and improve the reliability of the thrust chamber liners for both manned and unmanned space engine requirements, NASA has looked at numerous high-strength alloys, nonmetallic coatings and metallic protective coatings (ref. 3). The effort reported herein, however, is the first study of a composite material that attempts to limit the effect of large thermal strains on rocket engine life. This report describes the fabrication of the composite-lined chamber, the hot-firing test procedure and results, and the failure analysis of the liners.

Background

Because of NASA's need for high-temperature, high-pressure, reusable thrust chambers for numerous advanced rocket engines, the Lewis Research Center developed a program to find a suitable material for these thrust chamber liners. Early analyses (refs. 2 and 3) show that high pressures result in extreme heat fluxes of 8.8 to 29.4 kW/cm^2 deg C (30 to 100 Btu/in.^{2} sec). If chamber wall temperatures were to be reduced, chambers would have to be made from a high-conductivity material such as copper and be cooled regeneratively. In addition, the analyses showed that at these conditions the chamber wall, especially at its throat area, would have to withstand very large thermal strains. As part of these analyses, Lewis began its chamber materials program with a survey of high-conductivity alloys. From the list compiled, alloys were selected that had the best known combination of tensile and fatigue properties as well as high thermal conductivity. The selected alloys were tested for their elevated tensile and low-cycle fatigue properties and then were compared (ref. 2). Several of these alloys were chosen, together with Rocketdyne's Narloy Z, for fabricating subscale cylindrical thrust chambers (ref. 2) to be used in an actual rocket-firing environment.

The tests described in reference 2 indicate that Narloy Z and a NASA copper-silver-zirconium alloy have the longest and most uniform cyclic life under realistic thrust chamber conditions. However, a high-conductivity copper-tungsten composite developed at Lewis (ref. 4) appears to have a greater potential for accommodating high strains and extending thrust chamber life. Therefore, a program was undertaken to produce tungsten-wire-reinforced copper composite liners for testing in similar cylindrical thrust chambers to determine the life of the material.

Apparatus and Procedure

Liner and Chamber Fabrication

The development and fabrication of the composite liner were led by Leonard J. Westfall of Lewis (ref. 4). A removable mandrel was arc sprayed with Amzirc (a copper-zirconium alloy). This coated mandrel was hot isostatically pressed (HIPed) to densify the sprayed material and then machined to leave a 0.0584-mm (0.023-in.) thick copper alloy layer. This layer was then wound with 0.20-mm (0.008-in.) diameter tungsten wire and again arc sprayed with the Amzirc. Finally, the liner was HIPed again to densify the structure. Sufficient Amzirc was added to the outside so that cooling passages could be milled. Figures 1 and 2 show the liner on a mandrel and the embedded tungsten wire, respectively.

Figure 1.—Tungsten-wire-reinforced copper composite liner on mandrel.

The outside surface was then milled to form ribs and cooling passages. The cooling passages were filled with plater's wax and made conductive. Copper was electroform bonded to the ribs and allowed to build up to form the outer pressure jacket (fig. 3). Premachined copper inlet and outlet manifolds were assembled and electron beam welded to the liner-and-jacket assembly to complete fabrication of the subscale thrust chamber (fig. 4). Two similar chambers, designated here as chamber 1 (serial number 110) and chamber 2 (serial number 109), were prepared for testing.

Instrumentation

The instrumentation consisted primarily of thermocouples used to determine the hot-gas-side wall temperatures. A platinum resistance thermometer was installed in the linlet manifold to determine the inlet coolant temperature. An additional thermocouple was installed in the liquid hydrogen outlet manifold to determine the outlet coolant temperature. Pressure taps were used to determine the chamber operating pressures. In chamber 1, eight thermocouples were located in the thrust chamber throat plane, four thermocouples were located 1.49 cm (0.586 in.) upstream of the throat plane, and four thermocouples were located 1.27 cm (0.50 in.) downstream of the throat plane. The thermocouples were 0.025-mm (0.001-in.) diameter Chromel/constantan wires inside a sheath made of 0.356-mm (0.014-in.) diameter stainless steel tubing. The wires were spring-loaded and installed into 0.508-mm (0.020-in.) diameter drilled holes in the ribs to within 0.0889 mm (0.035 in.) of the hot-gas-side wall. Chamber 2 was instrumented like chamber 1 except that there were no thermocouples upstream or downstream of the throat. Figure 5 shows a completely instrumented subscale thrust chamber ready for testing.

Test Stand Assembly

Testing was conducted at the Lewis Research Center Rocket Engine Test Facility. This facility is a 222 410-N (50 000-lbf) sea-level rocket test stand equipped with an exhaust-gas muffler

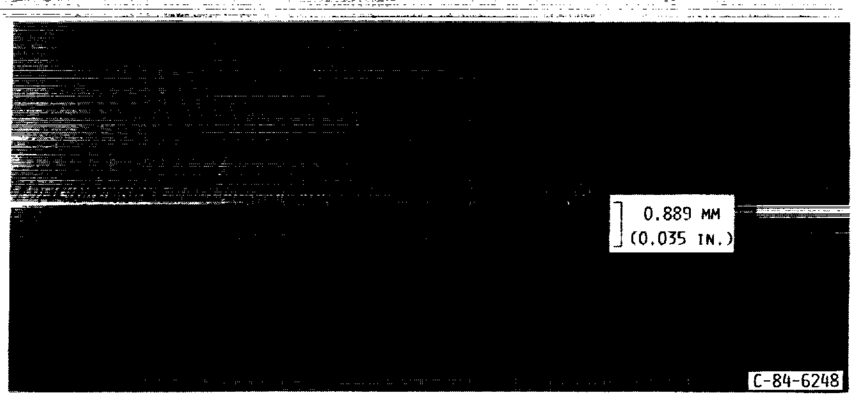

Figure 2.—Tungsten wire embedded in copper alloy.

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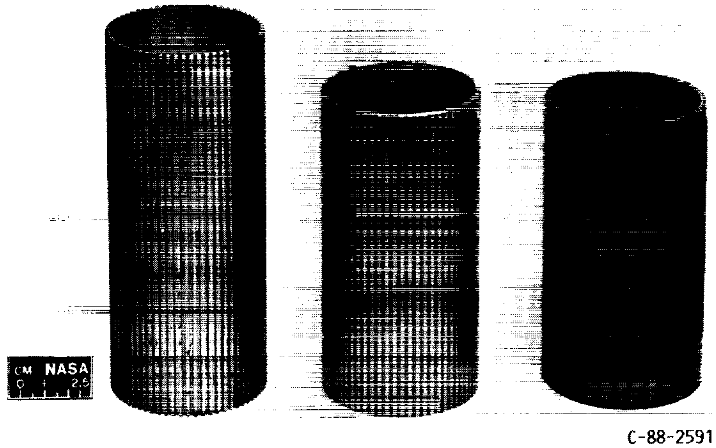

Figure 3.—Milled, filled, and closed-out thrust chamber liners, as shown from left to right.

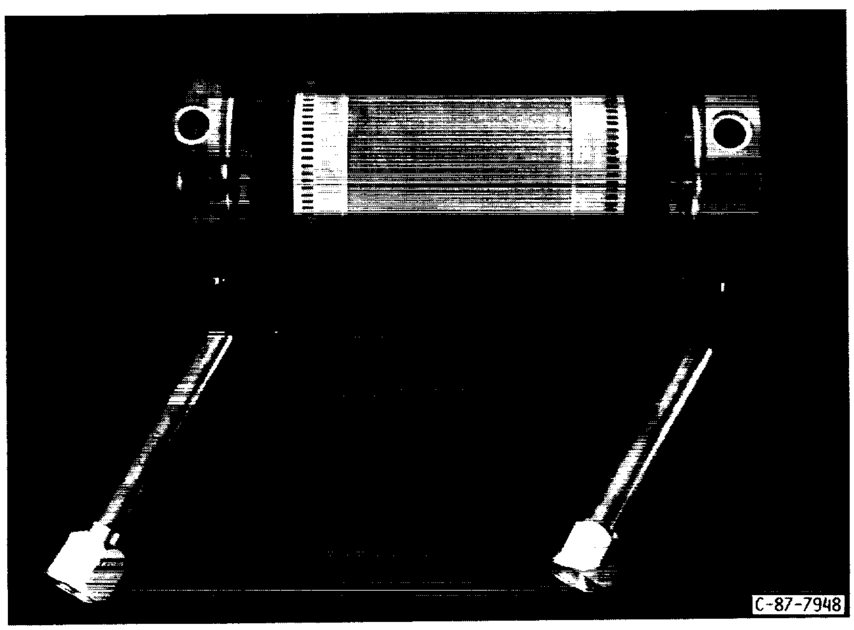

Figure 4.—Subscale thrust chamber assembly with inlet and outlet manifolds.

and scrubber. Pressurized storage tanks at the facility were used to supply liquid oxygen and ambient-temperature gaseous hydrogen propellants to the thrust chamber. Liquid hydrogen used to cool the cylindrical test section was disposed of through a burnoff stack. The thrust chamber exhaust gas and the water used to cool the centerbody were discharged into the scrubber. Because of the small volume of the thrust chamber combustion zone, an external spark torch igniter was used to backlight the oxygen and hydrogen propellants. Figure 6 shows a thrust chamber installed in the test stand during a test firing.

Hot-Fire Testing

Hot-fire testing consisted of burning gaseous hydrogen and liquid oxygen, together with a constant flow of high-pressure water through the centerbody and a constant flow of liquid hydrogen in the chamber cooling passages. The ignited propellants were allowed to burn 1.7 sec, at which time a steady-state temperature was reached. Then the propellant valves were shut off, and the chamber was allowed to cool down for 1.8 sec to bring the chamber back to liquid hydrogen temperatures. These heatup and cooldown steps were considered one complete test cycle and were repeated until the first sign of liner failure or leakage. As part of the normal procedure, the tests were monitored by a closed-circuit television camera and a test cell microphone. Both audio and visual outputs were recorded for playback. The chamber was assumed to have failed when the first through-the-wall crack had developed. With this type of failure a sudden and distinct high-pitched

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Figure 5.—Instrumented thrust chamber.

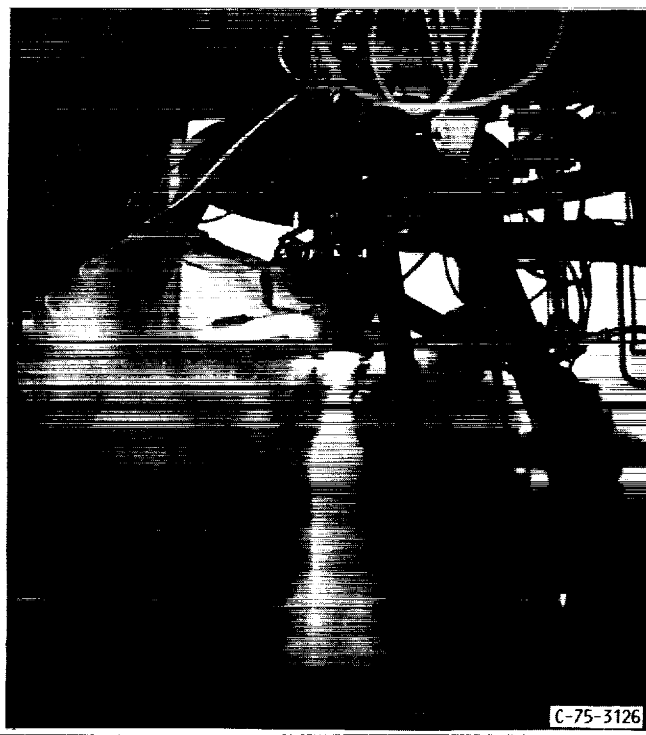


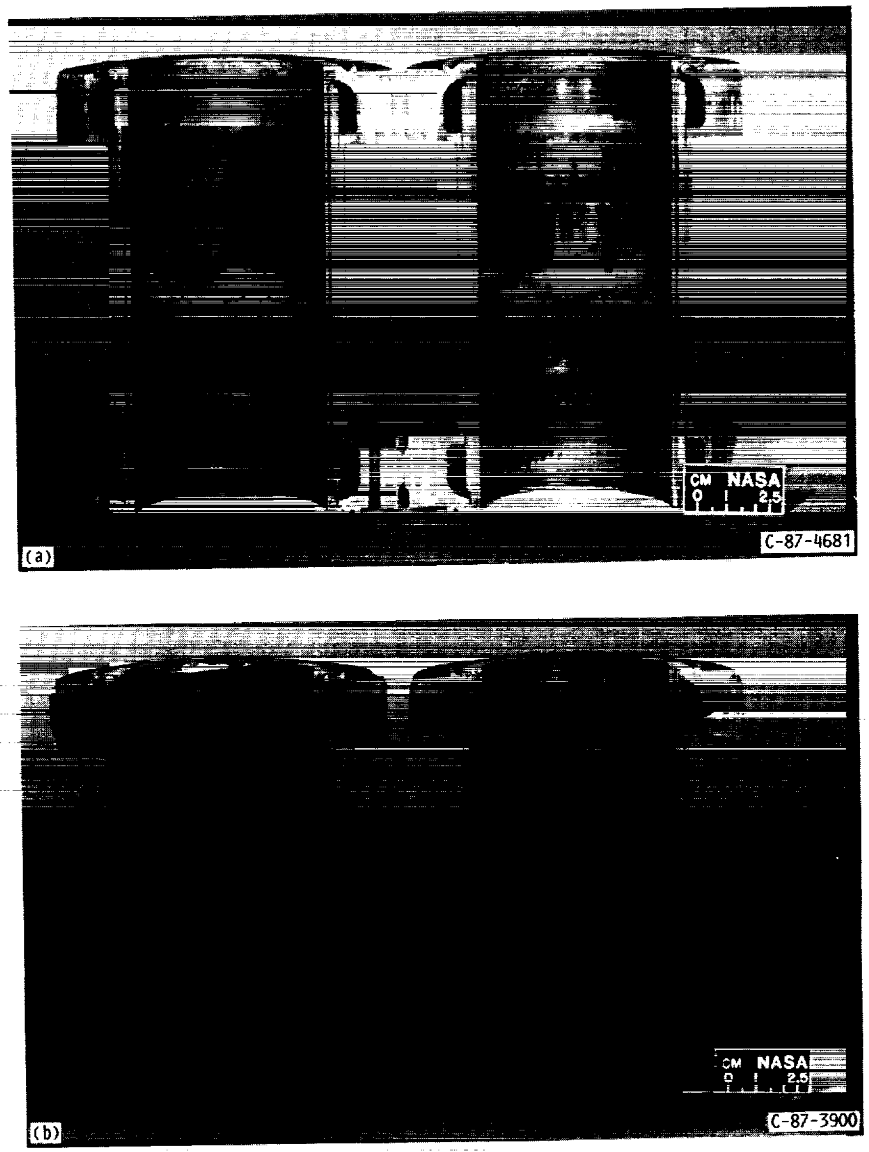
Figure 6.—Thrust chamber installed in test stand during hot firing.

hissing sound can be heard, and a vapor streak can be seen skewed to one side of the thrust chamber exhaust during the cooldown, or engine-off, portion of the cycle (refs. 2 and 3). For more information about the testing operation, including flows, temperatures, and pressures, see reference 2.

Chamber 1 showed no indication of failure until 115 test cycles were completed. Visual examination and leak checks revealed a circumferential crack located 1.49 cm (0.586 in.) above the throat plane. This type of failure and its location were unusual, since the highest temperatures, and hence the highest strains, always occur at the throat plane and since previous tests had shown that failures are always longitudinal and at the center of a channel. Chamber 2 was then installed in the test stand and tested for 400 cycles before it failed. Visual examination and leak checks showed the presence of three through-the-wall cracks; all were fine, circumferential (across rib and channel) cracks located in the throat plane, where the highest wall temperatures are experienced.

Metallurgical Evaluation

Prior to any metallurgical examinations, the chambers were carefully cut in half longitudinally and diagramed (fig. 7) to



(a) Chamber 1.
(b) Chamber 2.

Figure 7.—Halved test chambers after testing.

clearly define failure locations before any specimen was removed. The specimens in all cases were removed by wire electrical discharge machining to eliminate any possibility of disturbing the as-tested structure. After removal, all the selected specimens were mounted, polished, and photomicrographed. Some of the specimens were swabbed for 5 to 10 sec with a standard copper etchant consisting of 2 g of $K_2Cr_2O_7$, 8 ml of concentrated H_2SO_4, and four drops of HCl in 100 ml of water.

Although a large number of specimens were photographed to ensure that all aspects of the structures would be examined, only the significant photomicrographs are included and described in this report.

Discussion of Results

The results for each chamber, although similar, are discussed separately. Chamber 1, which was the first to be fabricated and tested, had the greatest number of thermocouples installed for a complete thermal mapping of the chamber hot-gas-side wall (eight evenly spaced thermocouples located in the throat plane, four located 1.49 cm (0.586 in.) upstream, and four located 1.27 cm (0.50 in.) downstream of the throat). The single failure in the chamber was located in the upstream thermocouple plane exactly between two thermocouples. This through-the-wall crack is shown in figure 8 from the inside chamber wall view.

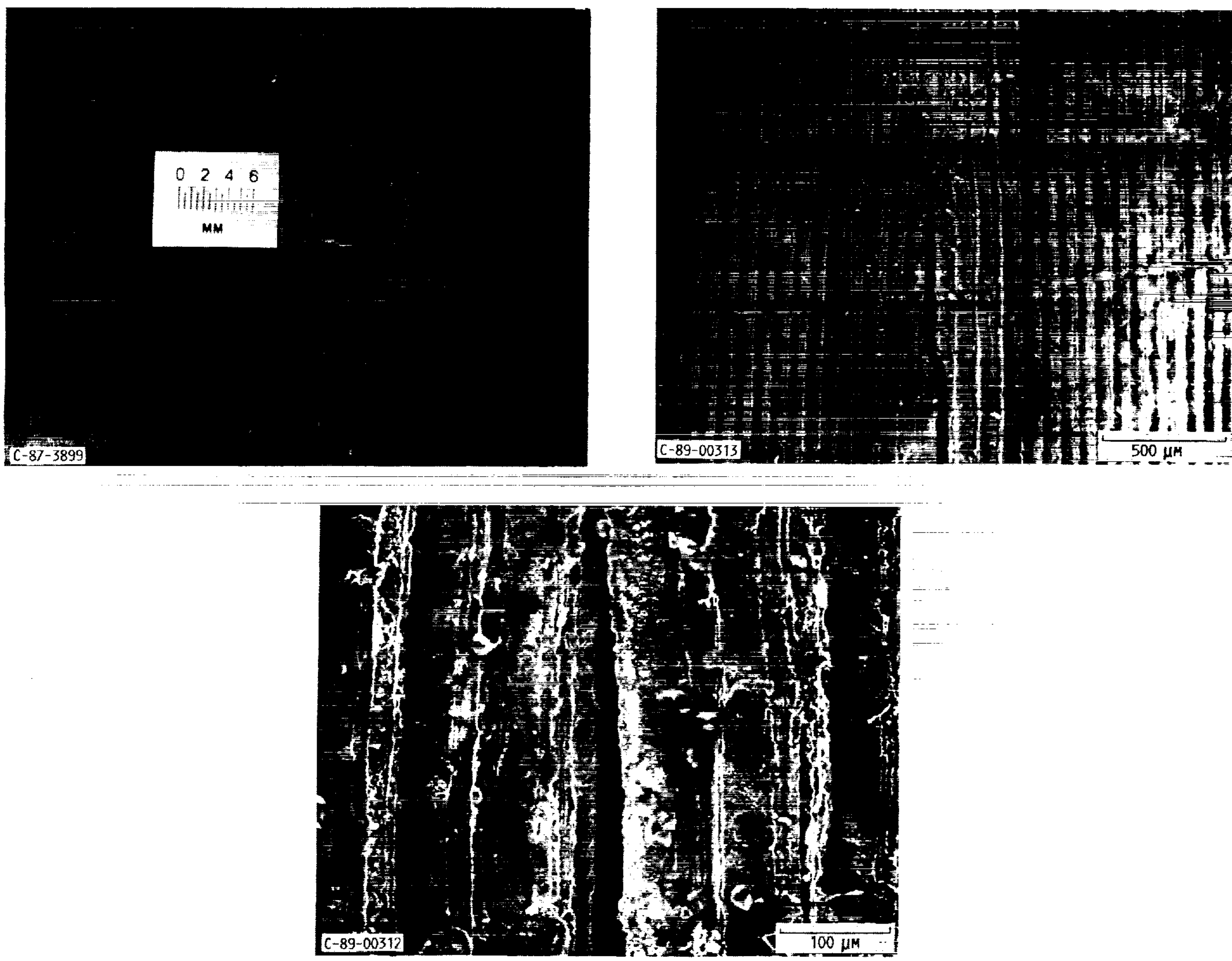


Figure 8.—Inside chamber wall view of through-the-wall crack in chamber 1.

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Also visible in figure 8 are the grooves caused by the final machining of the inside surface of the chamber. Both chambers had identical surface finishes. Figure 9 is an unetched cross-sectional view of these machining grooves. Although they are not considered a major factor, these grooves contributed to the early failures by acting as crack initiators.

Figure 10(a), which shows the through-the-wall crack, also reveals that the tungsten wires were distorted and damaged at this location and that a large void was exposed above the damaged region. When the specimen was etched to reveal the grain structure and reexamined, electroplated material was found in the void area (figs. 10(a) and (b)). Since the void was so large and the wires were damaged, the drawings and the fabrication process were reviewed. It was discovered that a thermocouple hole was inadvertently drilled in the wrong location. Figure 10 shows that copper was deposited in the exposed hole and sealed up during final electroform closeout. When the thermocouple hole was drilled, the depth was beyond the 0.889-mm (0.035-in.) limit from the hot-gas-side wall, and the drill penetrated into the liner and the reinforcing wires, causing distortion and damage to the wires and leaving a wall thickness of only 0.381 mm (0.015 in.) instead of the 0.889 mm (0.035 in.) required. This damaged, thinner section was determined to be the crack initiation site. With each test cycle, the crack continued to spread in both directions, following a circumferential machining groove until it reached a channel where it fractured and caused hydrogen to leak into the chamber. This explains the early failure of this chamber, especially in a cooler region than the throat area, where failure was not expected. Once this was discovered the failure analysis of chamber 2 was begun.

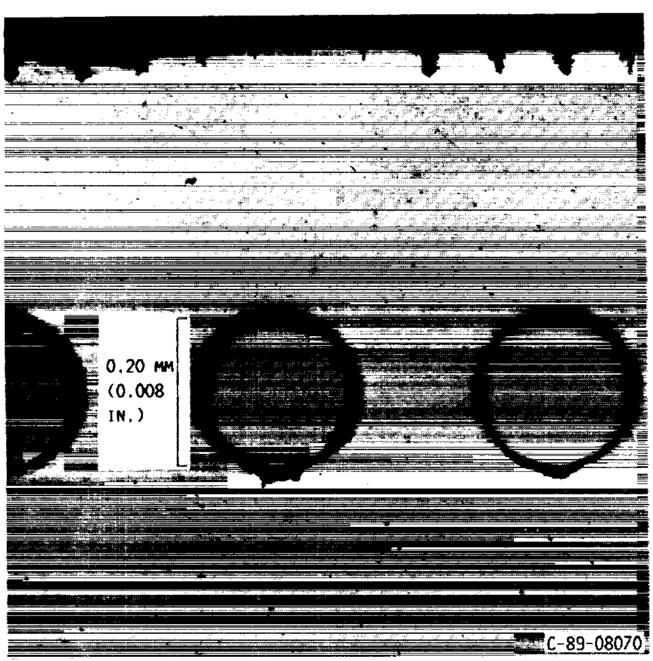
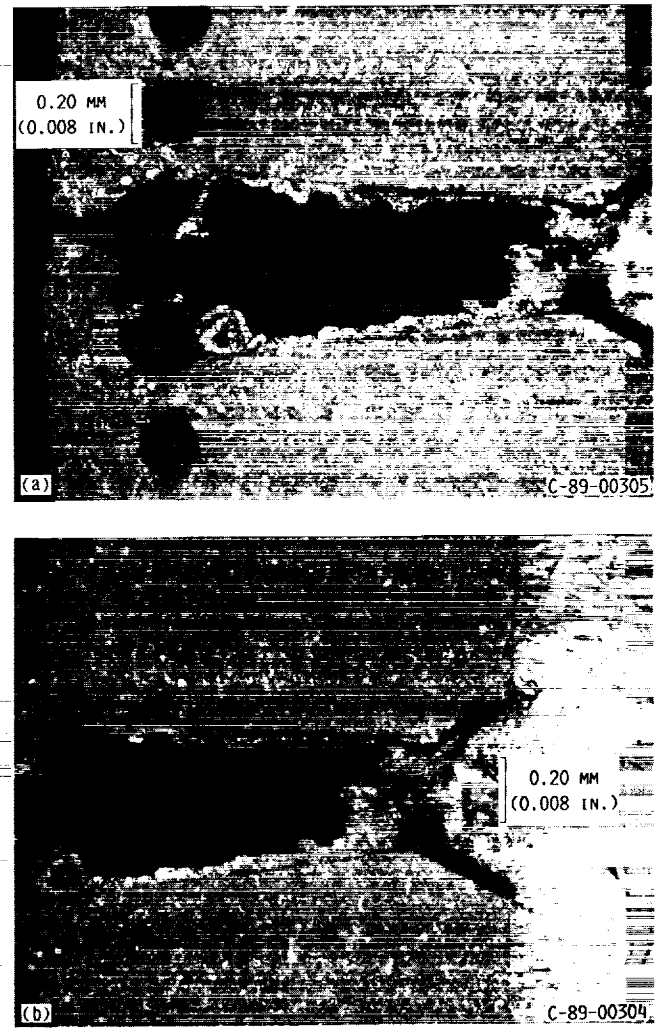


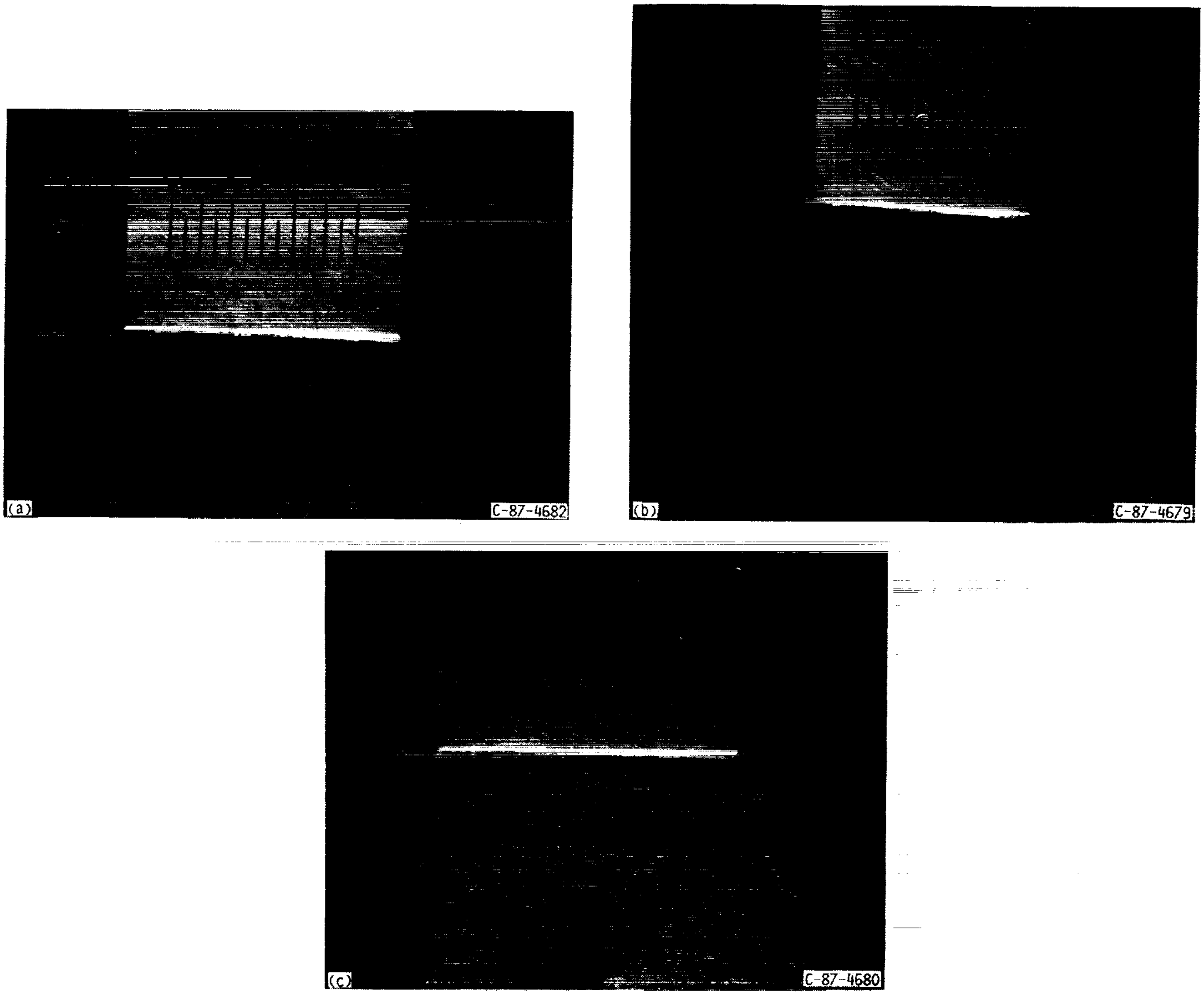
Figure 9.—Unetched cross-sectional view of machining grooves.



(a) Showing wire damage and large void.
(b) Showing copper deposited in thermocouple hole.

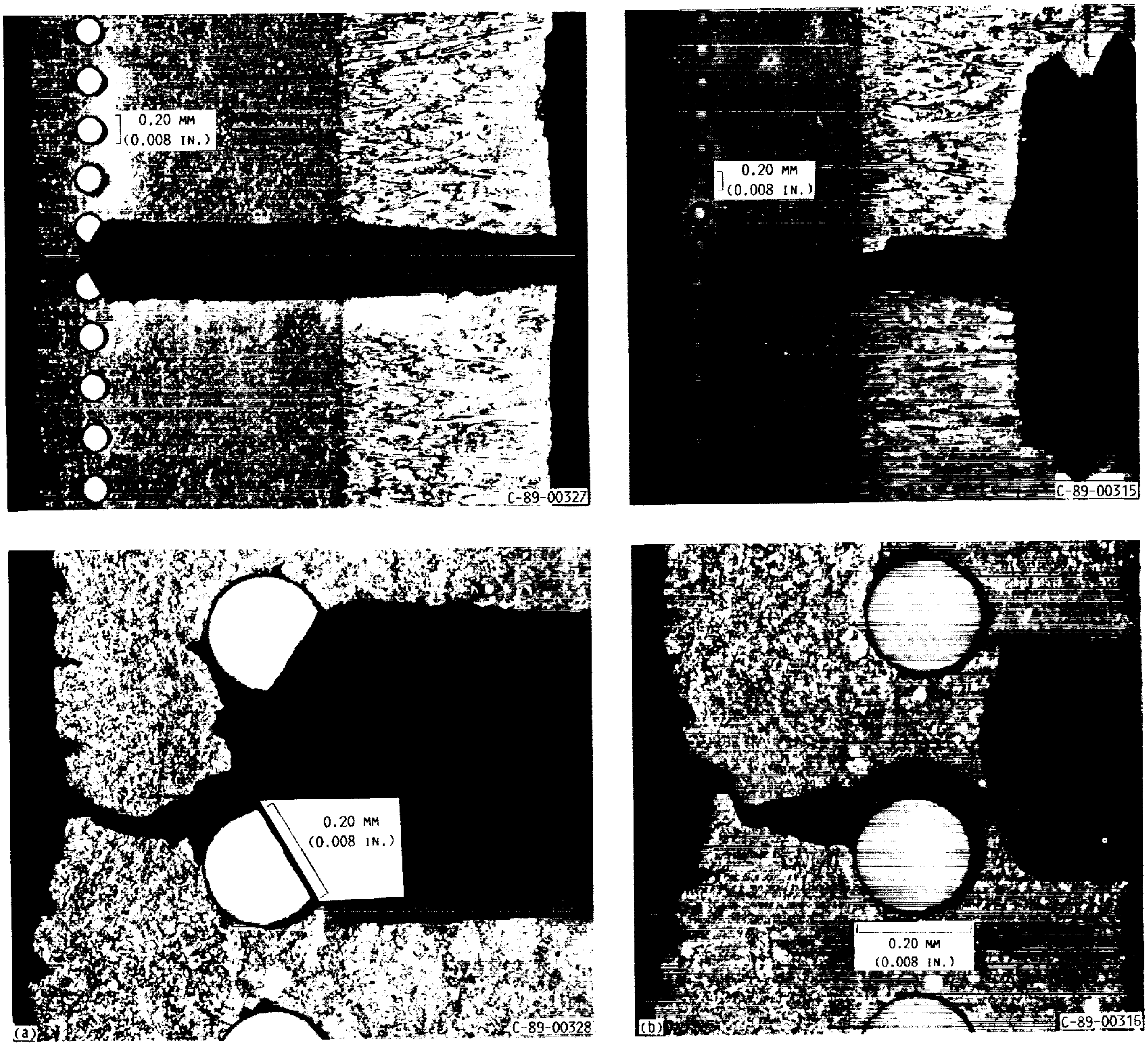
Figure 10.—Cross-sectional views of crack in chamber 1.

There were three crack locations on chamber 2, all in the throat plane. These cracks, designated here as cracks A, B, and C, are shown in figure 11. Sectioning and polishing revealed that all three crack sites were associated with a thermocouple hole. In all cases the holes were drilled to a depth greater than required and were the cause of early failure. Figure 12 shows cross-sectional views of cracks A and B. The views shown for crack C in figures 13 and 14 are different because this specimen was polished from the back side toward the wires rather than cross sectionally. Figure 13 shows the thermocouple hole and the crack leading from it in an area above the reinforcing wires, and figure 14 shows the damaged area after it had been polished further to expose the wires. There is no question that the cause of these failures was thermocouple hole penetration into the composite liner material.



(a) Crack A.
(b) Crack B.
(c) Crack C.

Figure 11.—Cracks in throat plane of chamber 2.



(a) Crack A.
(b) Crack B.

Figure 12.—Cross-sectional views of cracks A and B.

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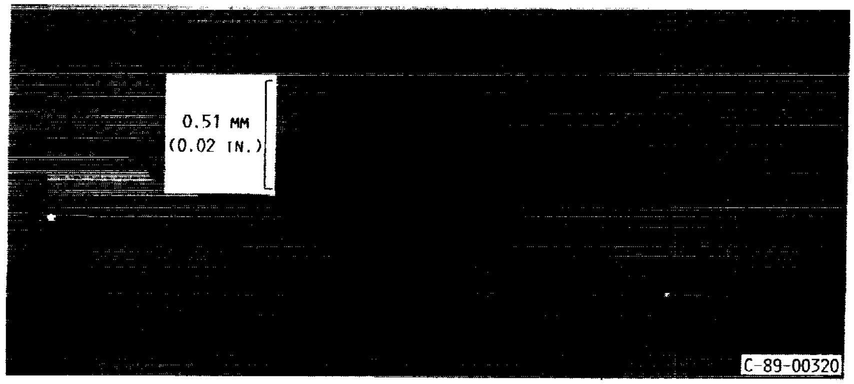

Figure 13.—Thermocouple hole and crack C.

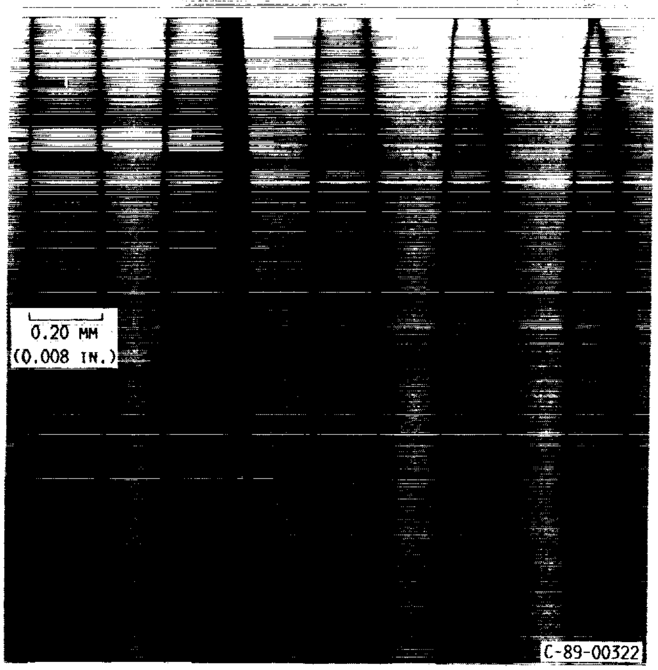

Figure 14.—Damaged area of crack C after further polishing.

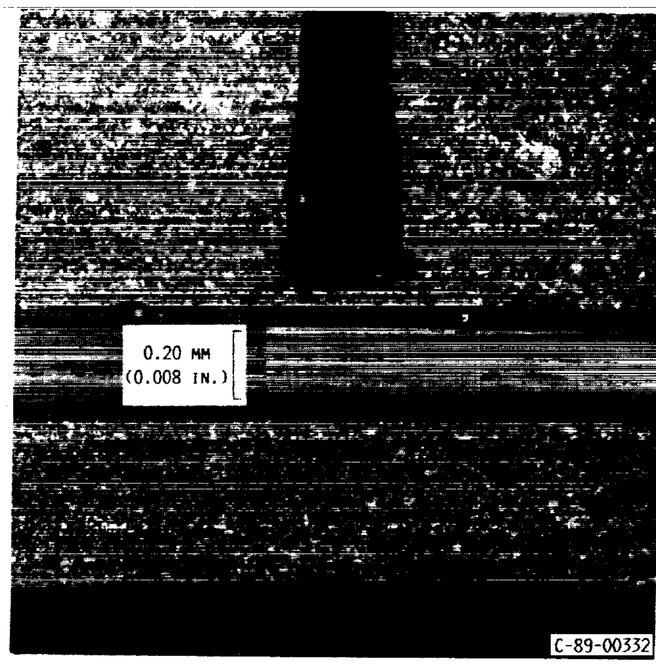

Figure 15.—Chamber 1 specimen from nonfailed thermocouple location.

Since a number of specimens were available, a sample from chamber 1 was prepared at a nonfailed thermocouple location (fig. 15). Here again the thermocouple hole was deeper than required, but since this location was not in the hotter throat plane, there was less thermal strain. Figure 15 also shows that the copper was not bonded to the tungsten wire, possibly indicating a limited amount of load sharing. This lack of bonding can be seen again in figure 16, cross-sectional views of a series of wires. The specimen shown in figure 16 was also taken from chamber 1 in a region remote from the throat area.

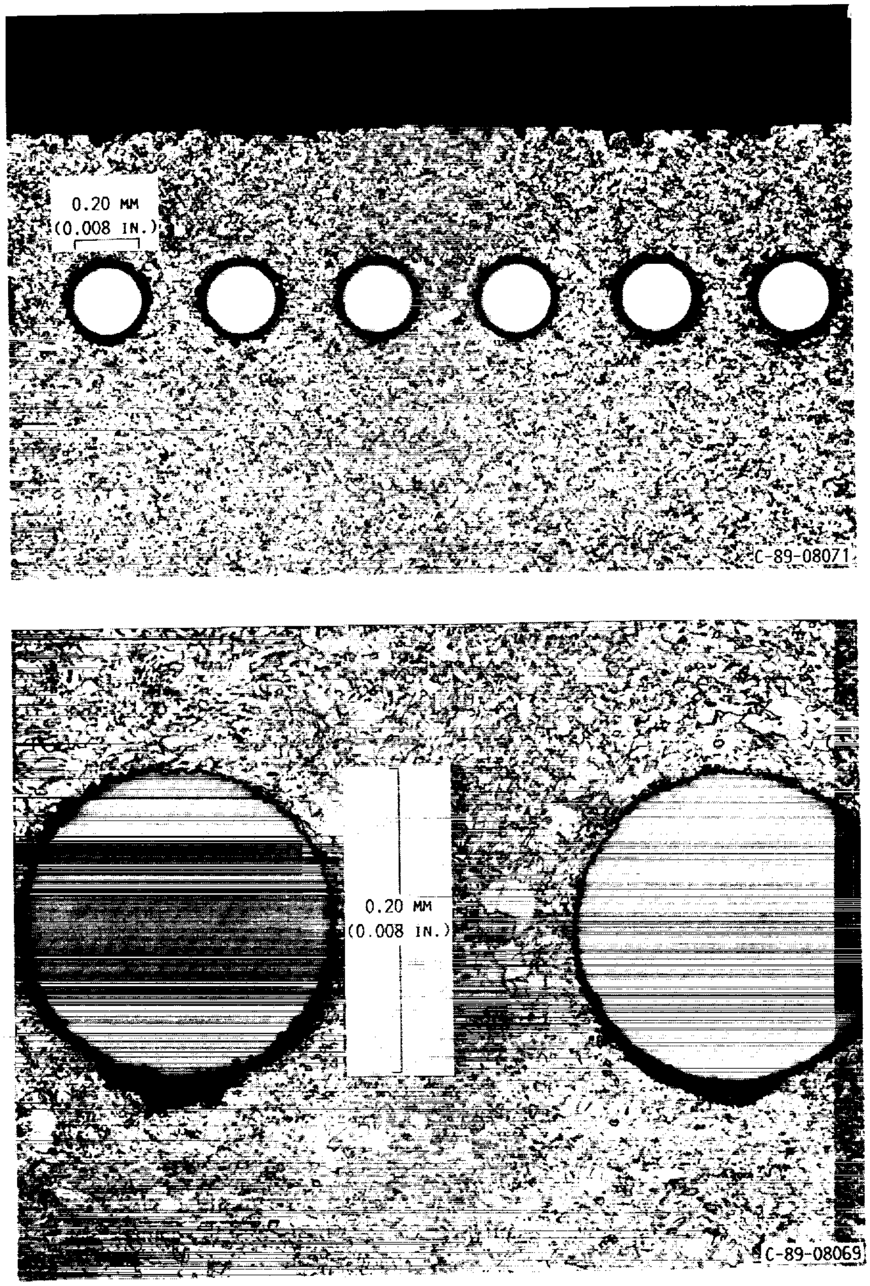

Figure 16.—Cross-sectional views of chamber 1 showing lack of bonding between tungsten wires and copper alloy.

In both chambers 1 and 2, the mode of crack propagation was very different than in past research. In the past, the failure was always longitudinal and at the center of a channel, as shown in figure 17. The direction of crack propagation in these tungsten-copper chambers was perpendicular to the channels (fig. 11(b)), 90° from all normal failure previously experienced. This new failure direction is assumed to be due to the load-carrying capability of the tungsten wires. The tungsten reinforcing wires carried the loads and resisted the strains in the hoop direction, thus causing the strain to accumulate in the copper alloy matrix between the wires.

Because the tungsten wires contained the strain in the hoop direction, the strain was relieved longitudinally instead of circumferentially. This 90° rotation of the strain made the thermocouple holes the thinnest, and weakest, points in the chamber liner. In the past, thermocouple depth was never a critical concern because experience had shown that the strain always accumulates in the longitudinal direction, thinning the channel and eventually cracking it. With this in mind, there was no evidence to warrant strict attention to the specified hole depth. In fact, it was thought that the closer the thermocouple bead was placed to the hot gases, the more accurate the

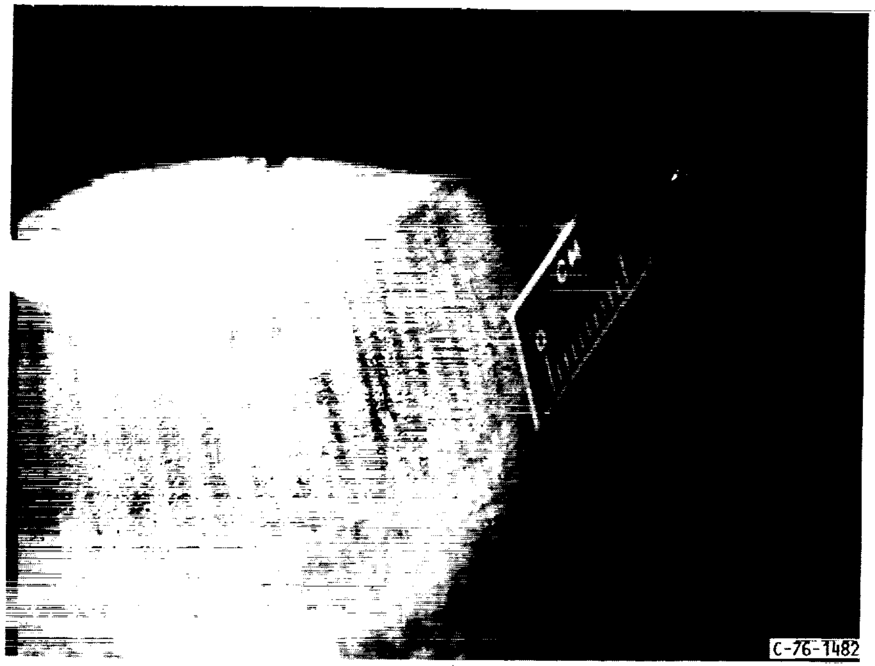

Figure 17.—Normal failure of thrust chamber showing longitudinal crack.

temperature reading would be. It has been shown here that with this type of metal matrix liner material the strains do not act in the same direction as previously experienced, and thus the placement of any type of instrumentation is of great concern.

Summary of Results

This work is part of NASA's continuing effort to extend the life of high-temperature, high-pressure rocket thrust chambers. Presently, these chambers are fabricated from a high-thermal-conductivity material such as copper. The hot-firing tests described herein were the first to incorporate the use of a composite liner material. The subscale rocket thrust chambers that were tested were lined with tungsten-wire-reinforced copper. The results of the tests were as follows:

1. Thermocouple holes were drilled too close to the hot-gas-side wall and thus contributed to early cracking.
2. Machining marks on the inside surface of the liner also contributed to the failure.
3. All failures were perpendicular to the channels, 90° from all normal failures previously experienced.
4. The reinforcing wires carried the loads and resisted the strains in the hoop direction. This caused strain accumulation in the copper alloy matrix between the wires.
5. There was evidence of a lack of bonding or adhesion of the matrix to the tungsten wires.

Concluding Remarks

Because several factors contributed to premature failure of the liner, it is highly recommended that this work be repeated to determine the benefits of this unique reinforced liner material. The following recommendations are also made:

1. Repeat testing should be at the same operating test conditions, with careful attention paid to controlling these conditions for the necessary life comparisons.
2. In order to help control conditions, the channels should be made identical; and the cover, or closeout, structure should be uniformly applied so that channel height does not vary.
3. Thermocouple holes should either not be used or be moved back away from the liner material to eliminate them as possible early failure sites.
4. The tungsten reinforcing wires should be cleaned and precoated in order to enhance bonding with the copper alloy. Coatings such as plated silver or copper could be used to protect the surface of the tungsten wire during processing and would readily diffuse when HIPed.
5. Before testing, the liner should be polished to remove all machining marks.

Acknowledgments

The authors wish to express appreciation to Albert J. Pavli for his observations of electroform material and recognition of machining marks and to Todd A. Leonhardt for producing the many photomicrographs.

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15. Supplementary Notes

16. Abstract

This report describes an advanced thrust chamber liner material, a tungsten-wire-reinforced copper composite, for potential use in long-life, reusable rocket engines. By strengthening the chamber in the hoop direction, this material may prevent longitudinal cracking due to low-cycle fatigue, which has been observed in conventional homogeneous copper chambers, without reducing the high thermal conductivity that is essential when operating with high heat fluxes. The tungsten-wire-reinforced copper composite was used in two hydrogen-oxygen rocket thrust chambers to evaluate its performance as a reusable liner material. Test results show that both chambers failed prematurely by cracking but that the crack sites were perpendicular to the normal direction of cracking. These findings indicate a degree of success in containing the tremendous thermal strain associated with high-temperature rocket engines. In all cases, the composite liner failures were associated with drilled instrumentation ports, and no other damage or deformation was found.

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