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Thrust Chamber Thermal Barrier Coating Techniques

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THRUST CHAMBER THERMAL BARRIER COATING TECHNIQUES

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SUMMARY

Methods for applying thermal barrier coatings to the hot-gas side wall of rocket thrust chambers in order to significantly reduce the heat transfer in high heat flux regions has been the focus of technology efforts for many years. This paper describes a successful technique developed by the Lewis Research Center that starts with the coating on a mandrel and then builds the thrust chamber around it by electroforming appropriate materials. This results in a smooth coating with exceptional adherence, as has been demonstrated in hot fire rig tests. The low cycle fatigue life of chambers with coatings applied in this manner has been increased dramatically compared to uncoated chambers.

INTRODUCTION

High pressure, reusable rocket thrust chambers, such as the Space Shuttle Main Engine (SSME), encounter a progressive deformation and thinning of the cooling passage wall during engine operation. The deformation is caused by a large, thermally induced, plastic strain that occurs in the hot-gas side wall during each thermal cycle. This phenomenon is known as plastic ratcheting and, after numerous thermal cycles, causes cracks to form in the cooling passage wall.[1-6]

One method of reducing, or eliminating the plastic strain, is to apply a thermal barrier coating (TBC), such as zirconium-oxide (ZrO_2), to the combustion-side wall of the thrust chamber. The use of TBC's to increase thrust chamber life is not a new concept. And many experimental programs have been conducted to perfect TBC's for use on rocket engines.[7-12] However, their use has been limited due to the tendency of the TBC's to spall after repeated thermal cycles. This is caused by the inability of the relatively brittle TBC to absorb the compressive strain that occurs in the coating during engine operation.

Most of the experimental work with TBC's was done with coatings greater than 0.25 mm thick, which is too thick for use on high pressure rocket thrust chambers. The TBC thickness required for use on a rocket thrust chamber, where the throat heat flux is in the range of 80 to 160 MW/m^2, is 0.12 to 0.012 mm,[12] depending on the effective thermal conductivity of the coating. Experience has shown that the thinner the coating, the better is its ability to absorb the compressive strain and remain bonded to the substrate.[13] Therefore, the use of TBC's on high pressure rocket thrust chambers offers the potential of substantially increasing life.

The normal procedure of applying TBC's is to apply a bond coat, such as nickel-chromium or nickel-chromium-aluminum-yttrium, onto the inner wall of an already fabricated thrust chamber liner, and then plasma spray the TBC layer over the top. Problems associated with this method of application include oxidation of the substrate, especially copper, during the application of the bond coat, a residual compressive stress which develops in the ceramic coating after the coated chamber returns to room temperature, and a relatively rough coating surface which increases the hot-gas-side heat transfer coefficient during engine operation. All of these phenomenon are detrimental to coating endurance.

Another method of applying a TBC, which reduces or eliminates the phenomenon described above, is to fabricate the thrust chamber inside out (i.e., start with the TBC and fabricate the chamber liner around it). A method of fabricating coated chambers by this technique, called the "electroform pickup process," was developed by the Lewis Research Center.[14] The process consists of plasma spraying the TBC onto a mandrel, applying the bond coat to the TBC, and then electroforming the thrust chamber liner around the coating. A 0.203-mm-thick ZrO_2 TBC applied to a subscale rocket thrust chamber with this technique demonstrated excellent performance during cyclic testing.[15] However, there remained a need to evaluate thin TBC's with this technique for application on high pressure rocket thrust chambers.

At Lewis, there has been an ongoing program to develop techniques for applying thin TBC's to rocket thrust chambers, both by applying coatings to already fabricated chambers,[16] and by applying coatings by using the electroform pickup process. The objective of the effort described in this report was to evaluate the performance and durability of thin TBC's applied to rocket thrust chambers by the electroform pickup process. The results reported herein are for a nominally 0.076-mm ZrO_2 coating applied to a subscale annular rocket thrust chamber by the electroform pickup process. The chamber was cyclically tested at a 4.14-MPa chamber pressure with liquid oxygen and gaseous hydrogen as propellants, and liquid hydrogen as the coolant. The experimental results are compared with an identical chamber tested without the TBC applied to the inner wall.

APPARATUS AND TEST PROCEDURE

Annular Rocket Thrust Chamber Assembly

Figure 1 shows a schematic of the annular rocket thrust chamber. The apparatus consisted of an annular injector; a contoured centerbody, which formed the combustion chamber, throat, and nozzle sections of the thrust chamber; and an outer cylinder, which served as the test section. The thrust chamber had a contraction and expansion area ratio of 1.79. At a chamber pressure of 4.14 MPa the thrust was approximately 5.34 kN with gaseous hydrogen and liquid oxygen as propellants. The cylindrical test section was separately cooled with liquid hydrogen, and the centerbody was cooled with water. Figure 2 shows the thrust chamber assembly with a cutaway of the cylindrical test section.

Injector

The injector was designed to operate with liquid oxygen and gaseous hydrogen. The oxygen was injected through 70 showerhead tubes arranged in two circular rows, 36 in the inner row and 34 in the outer row. The tubes were made of 0.23-cm-o.d. stainless steel having a 0.03-cm-thick wall. Two chamber pressure taps were located in the outer row of oxidizer tubes. All of the gaseous hydrogen was injected through the porous Rigimesh face plate. The face plate was removable, so that it could be replaced if damaged. The characteristic exhaust velocity efficiency averaged 95 percent.

Centerbody

The contoured centerbody was fabricated from copper with 40 rectangular cooling passages running axially throughout its length. The diameter in the combustion zone was 4.06 cm and 5.33 cm at the throat. The centerbody was 15.24 cm in length with a 7.5° half-angle conical expansion section. It was inserted through the injector and bolted into place from the back side.

A 0.076- to 0.127-mm zirconium-oxide coating was applied by conventional flame spray techniques to the outside surface to reduce the heat load and prolong the centerbody life. Water entered the centerbody from behind the injector, passed through the cooling passages, and was dumped at the thrust chamber exit.

Cylindrical Test Section

The conventional method of fabricating high pressure rocket thrust chambers is to machine cooling channels into a copper alloy liner and then close out the cooling channels by electro-deposition. If a coating (TBC) is to be used, it is normally applied to the chamber after fabrication. However, the cylindrical test section used in this program was fabricated by starting with the TBC and then building the chamber around it.

The cylinder was 15.24 cm in length and had an inside diameter of 6.60 cm. The fabrication sequence is shown figure 3. The process was started by machining an aluminum mandrel to the desired cylindrical shape. The TBC, in this case, an yttria-stabilized zirconia, was plasma sprayed onto the mandrel to a nominal thickness of 0.076 mm. A nickel-chromium bond coat was sprayed over the zirconium-oxide to a nominal thickness of 0.025 mm. The bond coat served not only to protect the copper substrate, but also rendered the surface electrically conductive. The mandrel assembly was then placed in a copper sulfate bath where it was continuously rotated during the plating operation. After a sufficient layer of copper was plated to the surface, the assembly was removed from the plating bath and the copper surface was machined to the proper diameter. Following this operation, 72 constant area cooling channels, 0.169 cm wide and 0.127 cm high, were milled into the copper liner. After machining, the cooling channels were filled with wax and the assembly placed back into the plating bath to electro-deposit the copper closeout. The wax was then removed from the cooling channels and the aluminum mandrel removed chemically. Final machining was performed and the manifolds added. This method of fabrication produced a smooth and even TBC having a surface finish of 0.7 μm rms. Figure 4 shows a cross-section of the cylinder wall with the cooling channel dimensions.

Instrumentation

The instrumentation of the cylindrical test section consisted primarily of chromel/constantan thermocouples. Eight thermocouples were located in the cooling passage ribs, four at one depth and four at another depth. Eight thermocouples were also located on the cylinder backside wall. The thermocouples were equally spaced circumferentially and all were located at the thrust chamber throat plane. The rib thermocouples were spring loaded against the bottom of the rib holes while the backside thermocouples were peened into the surface. Figure 4 shows typical thermocouple locations and figure 5 shows a coated cylinder with the instrumentation installed. In addition to the instrumentation described above, the liquid hydrogen inlet and outlet temperatures and pressures were also measured.

Test Facility

The tests were conducted at the Lewis Research Center rocket engine test facility. This is a 222 410-N sea-level rocket test stand equipped with an exhaust-gas muffler and scrubber. Propellants and coolants are supplied to the test stand using pressurized tanks. The liquid hydrogen used to cool the cylindrical test section was disposed of through a burn-off stack. The thrust chamber exhaust gas and the centerbody cooling water were expelled into the scrubber. Because of the small volume of the thrust chamber combustion zone, an external torch using gaseous hydrogen and gaseous oxygen was used to back-light the combustion chamber.

Data Recording

All pressures and temperatures were recorded in digital form on a magnetic tape. The digital recording system was set at a basic rate of 2500 samples per second. After processing, all of the data and calculations performed on the data could be printed out on the control room terminal.

Test Procedure

The cyclic tests were conducted so that the heat-up portion of the cycle was long enough for the hot-gas side wall temperature to reach steady-state, and where the chill-down portion of the cycle was long enough to bring the entire cylinder back to liquid hydrogen temperature. Total cycle time was 3.5 sec, 1.7 sec of burn time and 1.8 sec of chill-down time.

In order to create the maximum thermal strain in the cylinder wall, the liquid hydrogen flowed continuously for the entire cyclic test series. During the first cycle of any given test, a liquid hydrogen precool was used to bring the entire cylinder to liquid hydrogen temperature prior to ignition of the propellants. Short tests, consisting of just two cycles, were made before a cyclic test series in order to establish the desired chamber pressure and liquid hydrogen weight flow. After the desired test conditions were achieved, the thrust chamber was continuously cycled until the supply of liquid hydrogen was depleted. Generally, 70 to 90 cycles could be achieved on one tank of hydrogen. Figure 6 shows a computer plot of chamber pressure and liquid hydrogen weight flow for a typical cycle. Figure 7 shows the annular thrust chamber during cyclic testing.

A controller was used to control mixture ratio and chamber pressure. The liquid hydrogen flow was controlled by tank pressure and valve position. Water flow for the centerbody was set to the maximum obtainable and flowed continuously throughout the cyclic tests. All testing was monitored by closed-circuit television and a test cell microphone. The television and audio outputs were recorded on magnetic tape. After each test series, the cylinder was inspected to observe the condition of the coating.

TEST RESULTS

The objective of the program was to thermally cycle the coated cylinder to evaluate the performance and durability of the coating and its effect on cylinder life. After 1450 cycles, the testing was terminated due to the large amount of liquid hydrogen being consumed. The coating sustained only minor damage and proved to be very effective.

Post-test Destructive Analysis

After the testing was terminated, the cylinder was sectioned to study the coating morphology and to determine the amount of damage to the wall of the cooling passages.

Figure 8 shows the two halves of the cylinder after longitudinal sectioning. One half of the cylinder was also sectioned at the throat plane to examine a cross section of the coating and cylinder wall in the high heat flux region. It can be seen that there was no coating loss in the throat region, but some delamination of the ceramic layer occurred in the low heat flux region of the combustion zone. However, this delamination did not extend down to the bond coat, indicating a good bond between the ceramic layer and the bond coat.

The entire coating surface exhibited a micro-crack morphology, with the surface texture of the zirconia varying from relatively smooth in the throat region, shown at 500x in figure 9, to a rougher surface in the combustion zone upstream of the throat, shown at 250x in figure 10. The micro-crack morphology for the throat region is shown in cross section in figure 11. These surface cracks act as stress relieving locations and tend to prevent the coating from spalling. This phenomenon has been observed on ceramic coated turbine blades and attempts have been made to precrack the coating to provide these stress relieving sites before testing.[17-18]

The cylinder was also sectioned in the region upstream of the throat, where the coating delaminated. The coating surface is shown in figure 12 at a point where the delamination occurred. Note the micro-crack morphology in the upper portion of the figure where the coating remained intact. This same location is shown in cross section in figure 13. It can be seen that a layer of ceramic remains adhered to the bond coat.

Delamination of the coating within the ceramic layer is typical of the coating failure mechanism when there is good adhesion between the ceramic and the bond coat.[15,19] However, if a sufficient layer of ceramic remains, the substrate may still be adequately protected. An identical chamber, with a 0.2 mm zirconium-oxide TBC, was tested at the same conditions.[15] Seventy percent of the ceramic layer delaminated to a uniform thickness of approximately one-third of the original thickness after 80 thermal cycles. However, there was no loss of the coating down to the bond coat or substrate, and the cylinder wall remained well protected for an additional 579 cycles with no further delamination. The most significant point is that the cylinder tested in this program had no coating loss at the throat after 1450 cycles, which indicates that the thinner coating is more strain tolerant.

The effectiveness of the coating is exhibited in figure 14. This is a cross section of the cylinder wall at the throat plane. The original geometry of the cooling passages is still intact, and there is no apparent damage to the wall. This is in contrast to a typical thrust chamber wall failure as shown in figure 15. This figure shows a throat plane cross section of an uncoated cylinder which had an original cooling passage geometry identical to the coated cylinder.[4] The liner of this cylinder was fabricated from 1/2-hard Amzirc (Cu-0.15%Zr), which has significantly more strength than electroformed copper. A crack developed in the wall of one of the cooling passages after 393 thermal cycles. The deformation and thinning of the wall of the cooling passages is typical of the damage caused by plastic ratcheting. This demonstrates the dramatic effect a TBC can have on thrust chamber life.

Thermal Analysis

A TBC is very effective in increasing the life of a thrust chamber because it substantially reduces the operating temperature of the metal substrate and the thermal strain in the wall. In order to quantify this effect, a thermal analysis was performed, using the experimental temperature measurements to generate a thermal profile of the cylinder wall. The following procedure was used to perform the analysis: (1) a thermal model of the cylinder wall was developed, (2) the measured rib and backside wall temperatures were plotted as a function of time, and (3) a two-dimensional conduction analysis was performed, using the SINDA thermal analyzer program,[20] until a best fit of the experimental temperature measurements could be achieved.

Figure 16 shows the model used for the thermal analysis. Because of symmetry, only one-half of a cooling channel wall cross section is required, see figure 4. Also shown on the model are the two depths at which rib thermocouples were located circumferentially around the cylinder, denoted as T.C.1 and T.C.2. The backside thermocouples are denoted as T.C.3.

Input to SINDA requires the hot-gas side and coolant side boundary conditions as a function of time and the thermal conductivity of the copper substrate, the bond coat, and the ceramic as a function of temperature. The steady-state hot-gas side heat transfer coefficients were obtained with a water cooled calorimeter for the annular thrust chamber configuration. The steady-state coolant-side heat transfer coefficients were calculated from empirical correlations which best describe convective heat transfer to liquid hydrogen. The hot-gas side and coolant side transient heat transfer coefficients were not available from experimental data; therefore, as an initial input, they were assumed to vary from their steady-state values in proportion to variations in chamber pressure and liquid hydrogen weight flow for each time slice. This procedure did a reasonable job of fitting the temperature data, although minor adjustments were made in the boundary conditions, where needed, to best fit the experimental data.

Since the measured parameters remained essentially the same throughout the cyclic testing, the data from any cycle was considered to be representative of the data from all cycles. Figure 17 shows a plot of the experimental temperature data from the 861st cycle of the coated cylinder. The data shown for T.C.1, T.C.2, and T.C.3 represent the averaged temperature measurements of the thermocouples at those locations. Also shown on this figure are the calculated matching temperatures of the experimental data at T.C.1, T.C.2, and T.C.3; the calculated temperature at the interface of the substrate and the bond coat; and the calculated hot-gas side wall temperature. Although not shown on this figure, the temperatures for every nodal point in the model were also calculated.

The reason a TBC can be so effective, especially on a high pressure rocket thrust chamber, is because a very large temperature drop can be achieved across the wall of a relatively thin coating. This is due to the high heat flux associated with a high pressure rocket thrust chamber and the extremely low thermal conductivity of the ceramic coating. This effect is readily apparent in figure 17. Although the steady-state hot-gas side wall temperature is nearly 2000 K, reflecting the higher operating temperature of the TBC, the temperature drop across the coating, which includes the bond coat, is nearly 1670 K. In order to show the effect the TBC has on the substrate wall temperatures, a comparison of wall temperatures is made with an identical uncoated copper cylinder tested at the same conditions.[4,15] The results are shown in table I. It can be seen that the copper substrate wall temperature was reduced from 844 K, which is the hot-gas side wall temperature for the uncoated cylinder, to 334 K for the coated cylinder. The temperature drop across the copper wall was reduced from 556 K to 194 K, respectively. These are substantial reductions in the cylinder wall operating temperatures. Since the tensile strength of the material varies inversely with temperature and the thermal strain varies directly with temperature difference, a

TBC can have a profound effect on the operating stresses and strains within a thrust chamber wall. This is apparent in the life of the two cylinders. The uncoated cylinder developed cracks in the wall after only 210 cycles, which was the typical life of cylinders with OFHC (oxygen-free high conductivity) copper liners.[4] The coated cylinder showed no damage to the wall after 1450 cycles, demonstrating that the effects of plastic racheting were eliminated as shown in figure 14.

CONCLUSIONS

The life of reusable, high pressure rocket thrust chambers can be significantly increased through the use of thermal barrier coatings (TBC's) on the chamber wall. A TBC can reduce the substrate wall temperatures and the thermal strain in the wall to the extent that the damage caused by cyclic plastic ratcheting can be virtually eliminated.

A TBC used in a high heat flux environment must be thin to keep the coating temperature within its operating limits. Based on the results of the fabrication and cyclic testing of an annular rocket thrust chamber, it has been demonstrated that a thin, durable TBC can be applied to a chamber wall by the electroform pickup process. This technique of applying TBC's needs to be validated on a high pressure, contoured rocket thrust chamber.

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Table I

Configuration	Hot-gas side wall temperature, T_{gw},[a] K	Copper substrate wall temperature at bond coat interface, T_{cw},[a,b] K	Backside wall temperature of cylinder, T_{bw},[a] K	Temperature difference across copper wall, $T_{cw} - T_{bw}$,[a] K	Number of cycles	Remarks
Coated cylinder[c] (Electroformed copper liner)	2000	334	140	194	1450	No damage to the wall of the cooling passages.
Uncoated cylinder[d] (OFHC copper liner)	844	844	288	556	210	Cracks in the wall of the cooling passages, severe deformation and thinning of the wall due to plastic ratcheting.

[a]Calculated temperature locations shown on fig. 16.
[b]Or the copper hot-gas side wall temperature of the uncoated cylinder.
[c]See fig. 17.
[d]See ref. 15.

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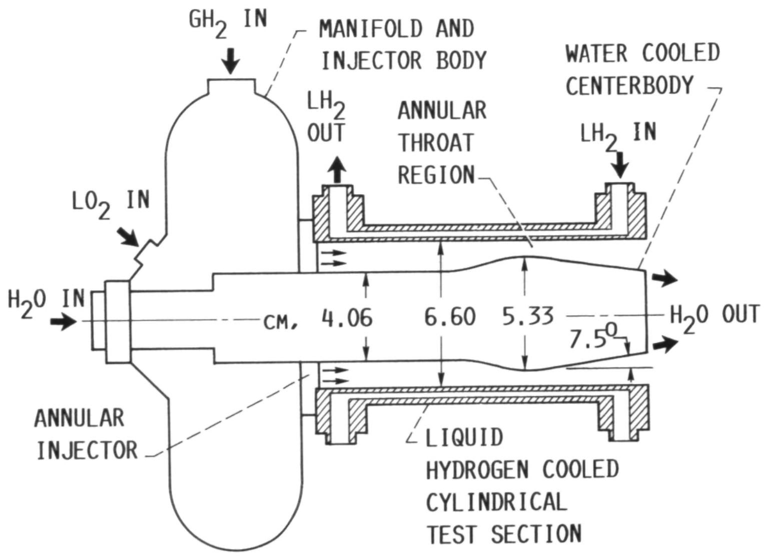

FIGURE 1. - SCHEMATIC OF ANNULAR ROCKET THRUST CHAMBER ASSEMBLY.

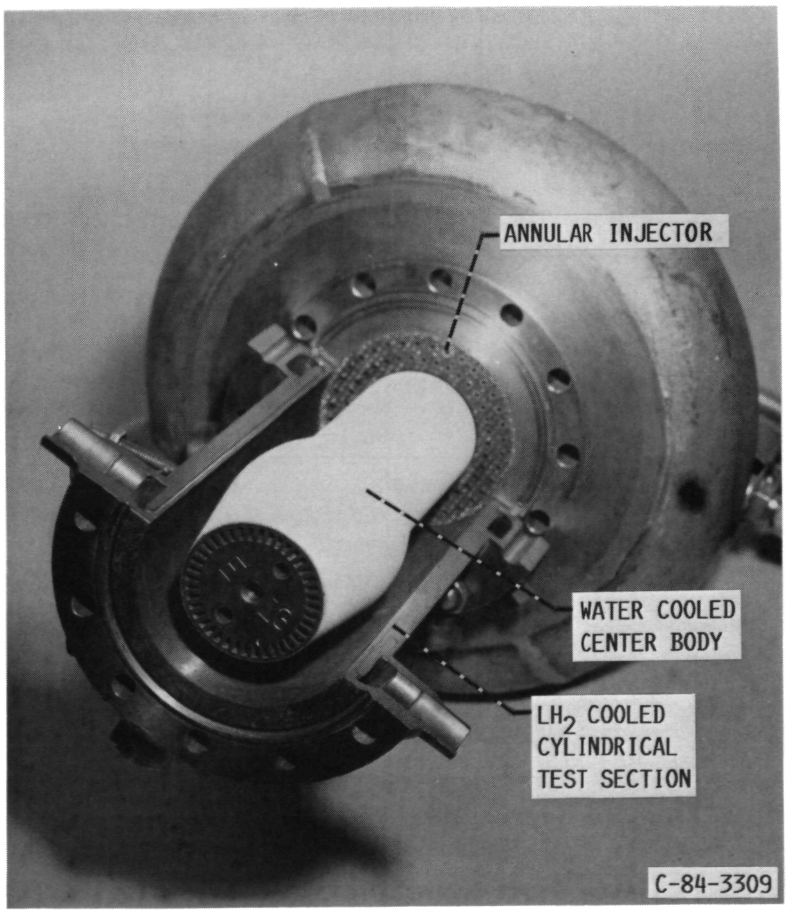

FIGURE 2. - ANNULAR THRUST CHAMBER ASSEMBLY.

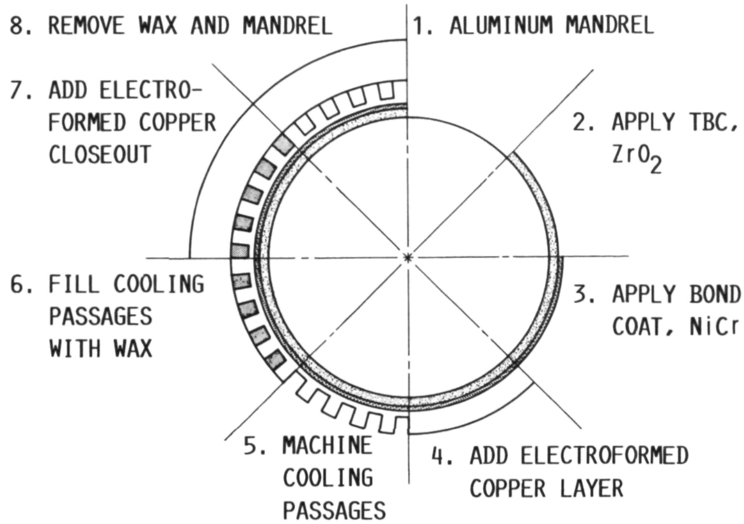

FIGURE 3. - FABRICATION SEQUENCE FOR COATED CYLINDERS.

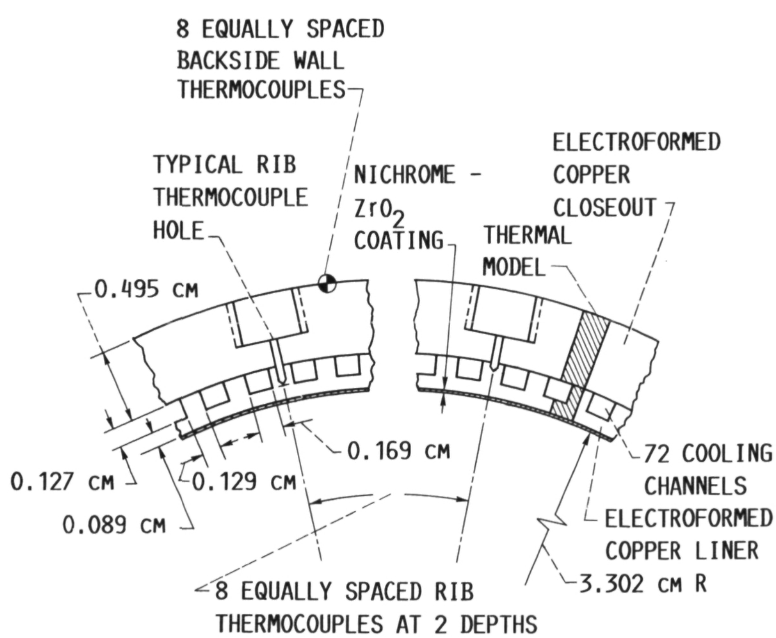

FIGURE 4. - CYLINDER WALL CROSS SECTION SHOWING INSTRUMENTATION LOCATIONS AND DIMENSIONS.

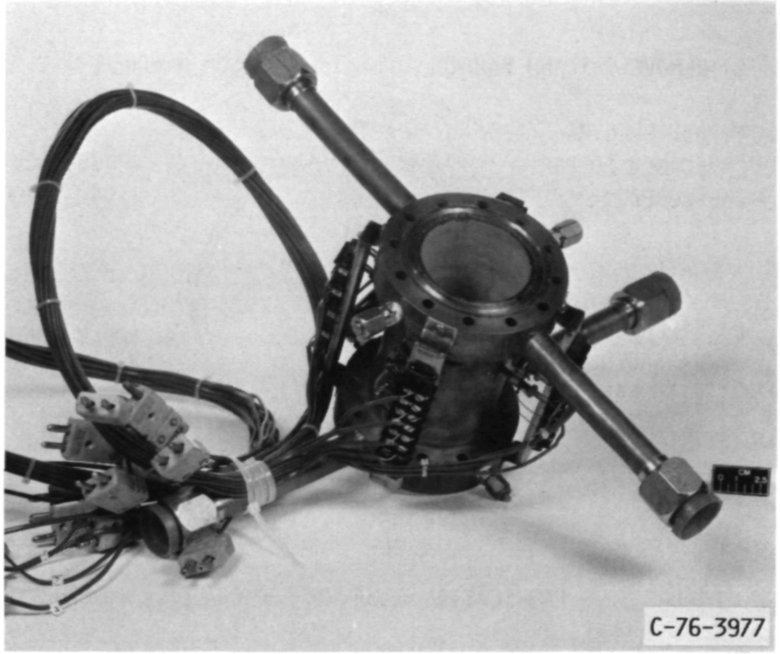

FIGURE 5. - COATED CYLINDRICAL TEST SECTION WITH INSTRUMENTATION INSTALLED.

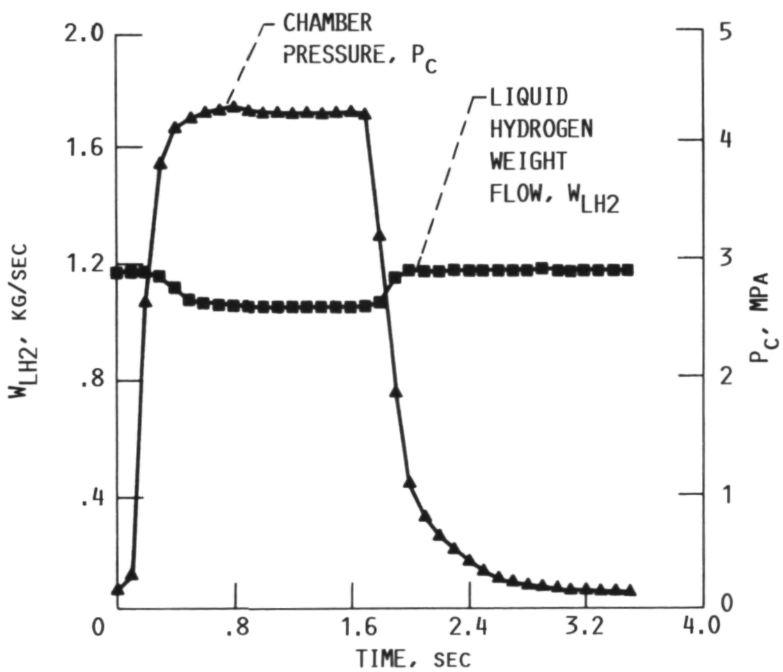

FIGURE 6. - COMPUTER PLOT OF EXPERIMENTAL CHAMBER PRESSURE AND LIQUID HYDROGEN WEIGHT FLOW FOR A TYPICAL CYCLE.

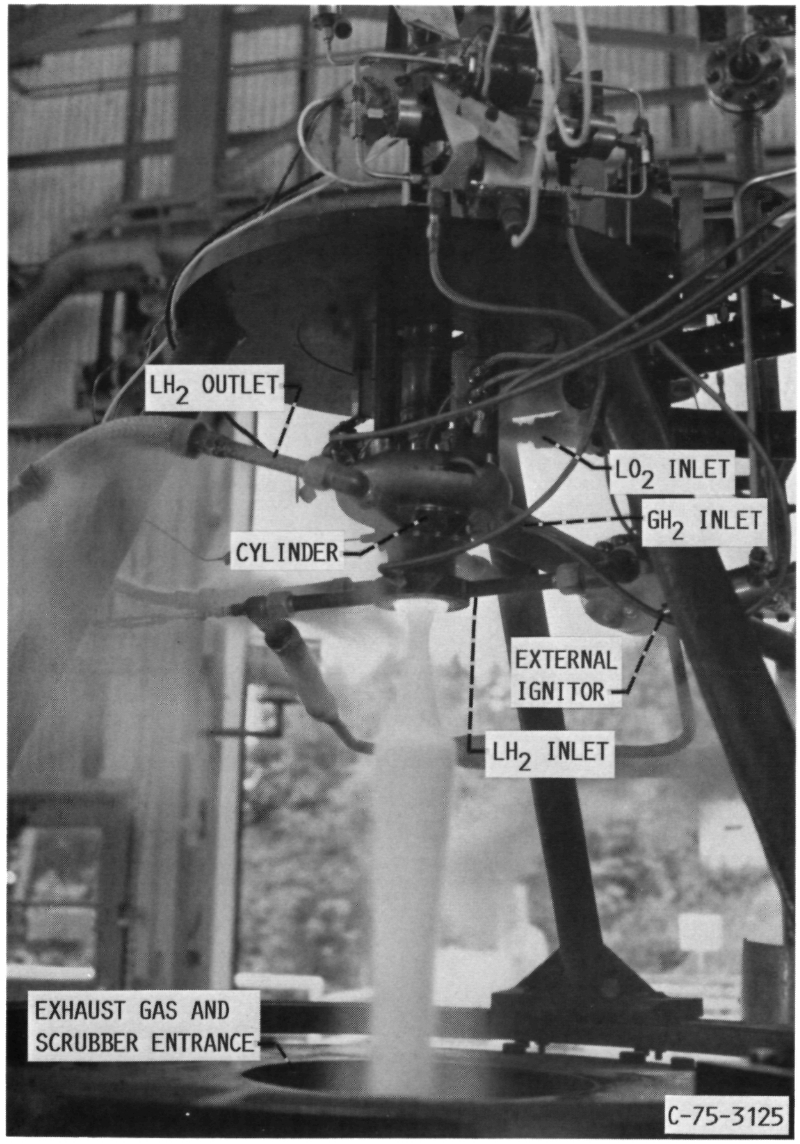

FIGURE 7. - ANNULAR THRUST CHAMBER DURING CYCLIC TEST.

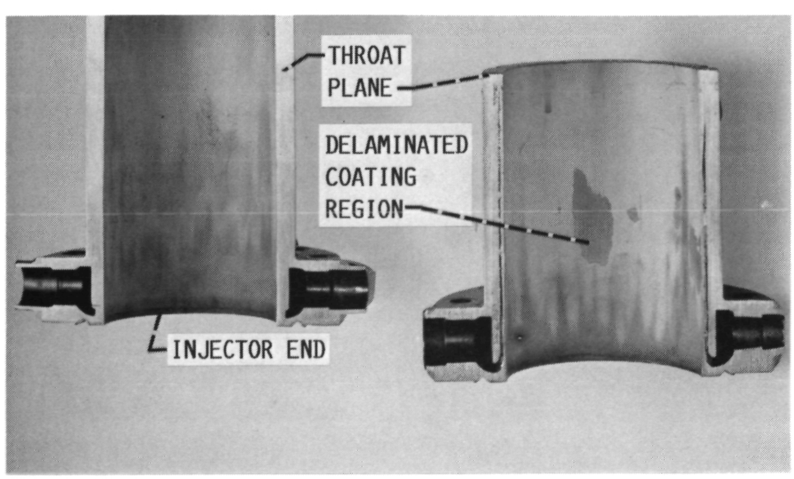

FIGURE 8. - POST-TEST SECTIONS OF CYLINDRICAL TEST SECTION.

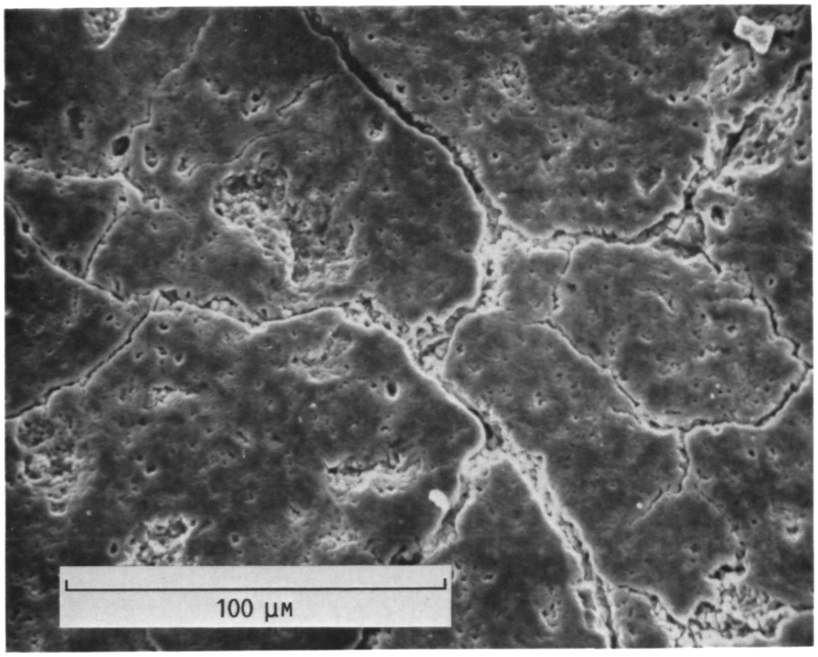

FIGURE 9. - MICRO-CRACK MORPHOLOGY IN THROAT REGION. 500x.

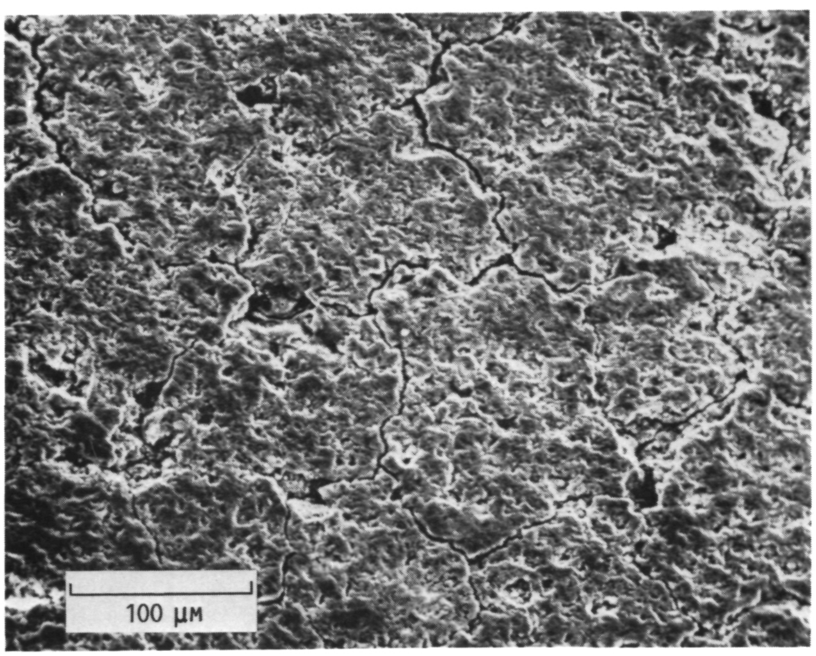

FIGURE 10. - MICRO-CRACK MORPHOLOGY IN COMBUSTION ZONE. 250x.

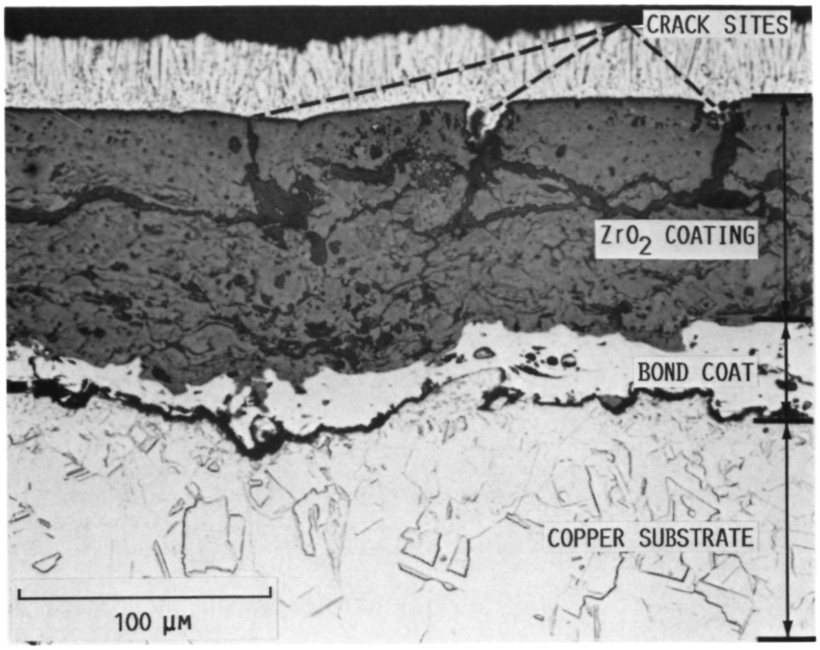

FIGURE 11. - CROSS SECTION OF CRACK MORPHOLOGY IN THROAT REGION. 400x.

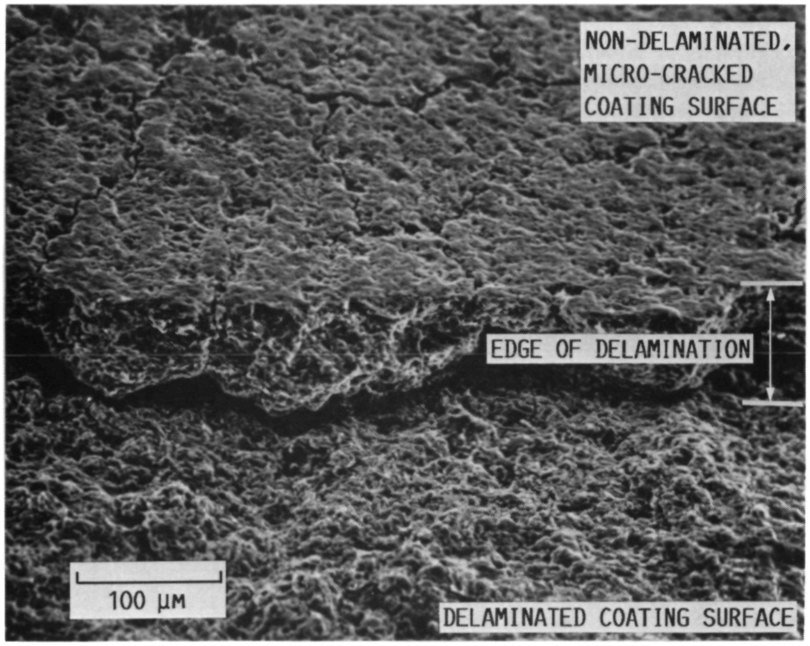

FIGURE 12. - OBLIQUE VIEW OF COATING SURFACE AT THE LOCATION OF DELAMINATION. 200x.

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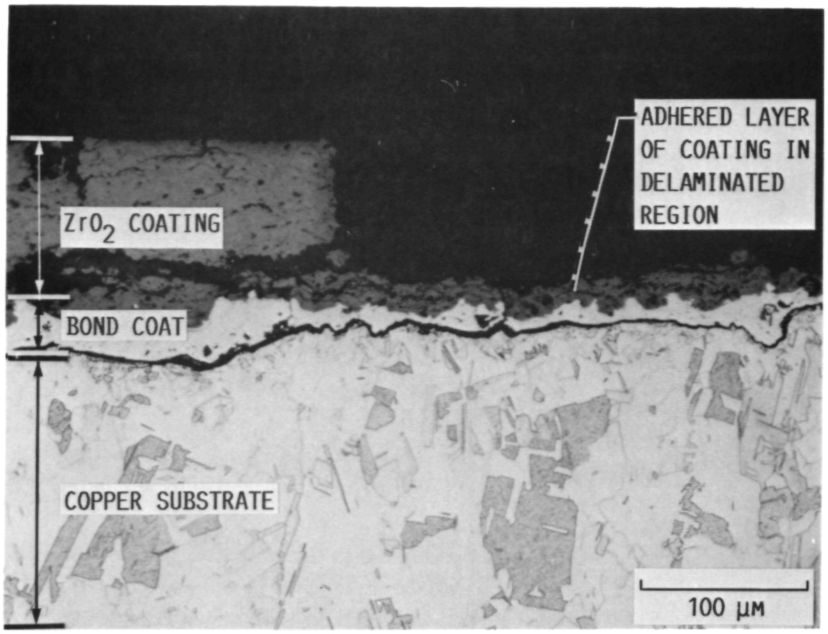

FIGURE 13. - CROSS SECTION OF COATING AT THE LOCATION OF DELAMINATION. 200x.

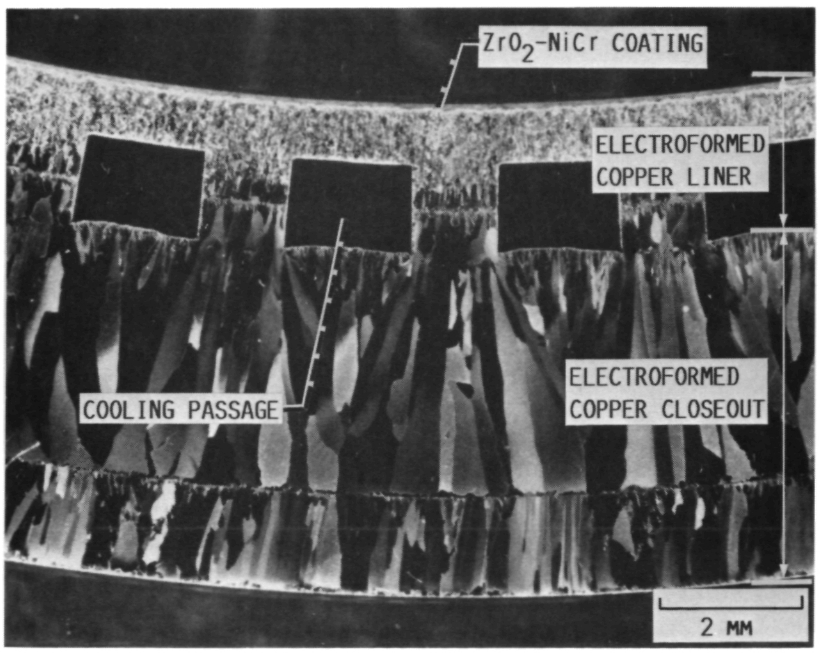

FIGURE 14. - CROSS SECTION OF COATED COPPER CYLINDER AT THE THROAT PLANE AFTER 1450 THERMAL CYCLES. 10x.

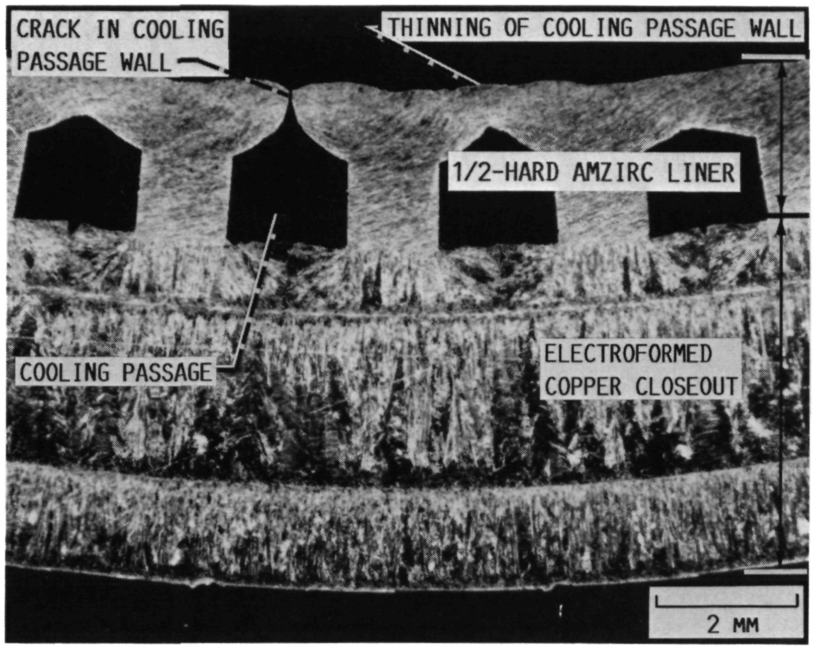

FIGURE 15. - CROSS SECTION OF 1/2-HARD AMZIRC CYLINDER AT THE THROAT PLANE AFTER 393 THERMAL CYCLES. 10x.

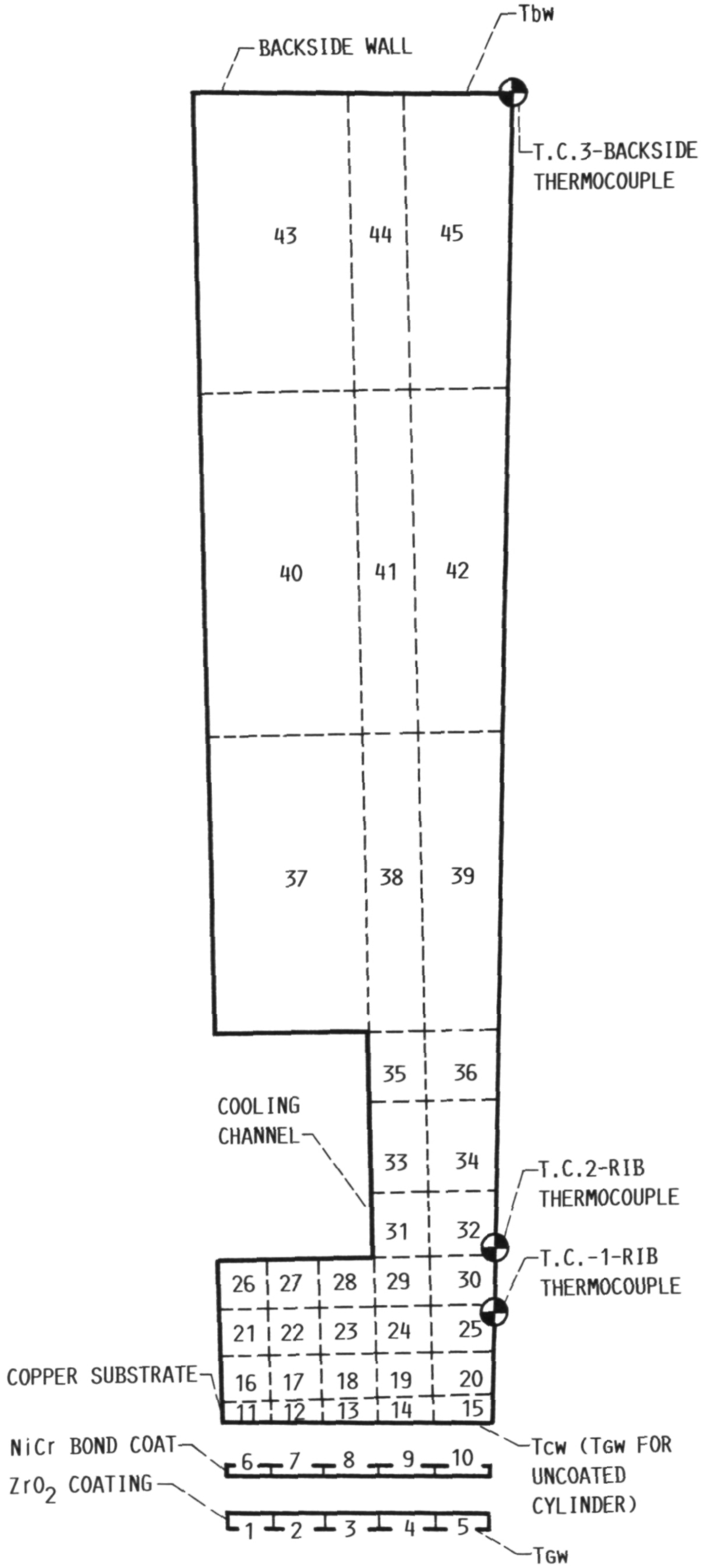

FIGURE 16. - 45 ELEMENT THERMAL MODEL.

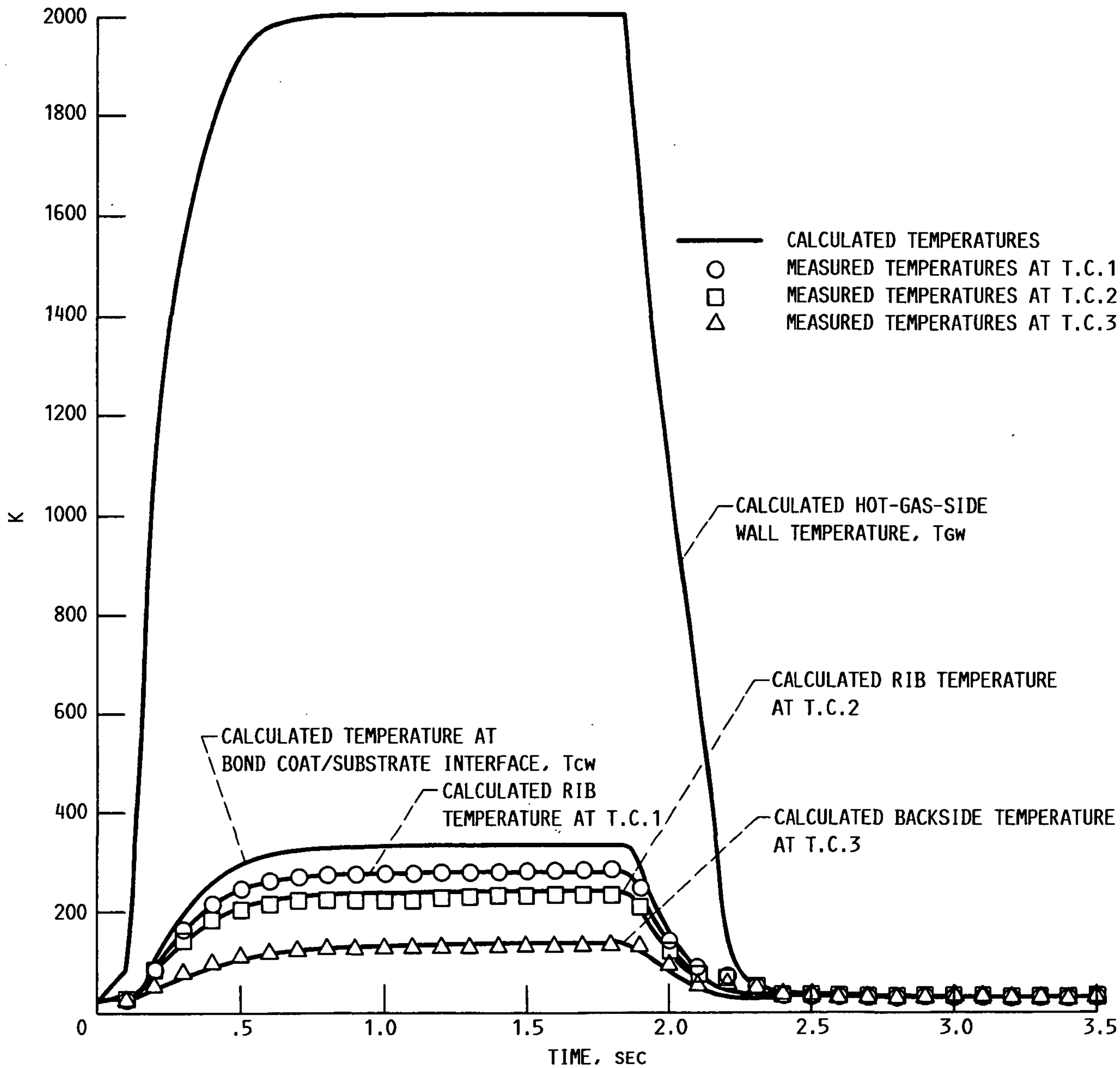

FIGURE 17. - MATCHING OF CALCULATED TEMPERATURES WITH EXPERIMENTAL TEMPERATURE MEASUREMENTS.

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16. Abstract

Methods for applying thermal barrier coatings to the hot-gas side wall of rocket thrust chambers in order to significantly reduce the heat transfer in high heat flux regions has been the focus of technology efforts for many years. This paper describes a successful technique developed by the Lewis Research Center that starts with the coating on a mandrel and then builds the thrust chamber around it by electroforming appropriate materials. This results in a smooth coating with exceptional adherence, as has been demonstrated in hot fire rig tests. The low cycle fatigue life of chambers with coatings applied in this manner has been increased dramatically compared to uncoated chambers.

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