

ROCKET THRUST CHAMBER THERMAL BARRIER COATINGS

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Abstract

Subscale rocket thrust chamber tests were conducted to evaluate the effectiveness and durability of thin yttria stabilized zirconium oxide coatings applied to the thrust chamber hot-gas side wall. The fabrication consisted of arc plasma spraying the ceramic coating and bond coat onto a mandrell and then electrodepositing the copper thrust chamber wall around the coating. The remaining fabrication was completed in a conventional manner. Chambers were fabricated with coatings .008, .005 and .003 inches thick. The chambers were thermally cycled at a chamber pressure of 600 psia using oxygen-hydrogen as propellants and liquid hydrogen as the coolant. The thicker coatings tended to delaminate, early in the cyclic testing, down to a uniform sublayer which remained well adhered during the remaining cycles. Two chambers with .003 inch coatings were subjected to 1500 thermal cycles with no coating loss in the throat region, which represents a tenfold increase in life over identical chambers having no coatings. An analysis is presented which shows that the heat lost to the coolant due to the coating, in a rocket thrust chamber design having a coating only in the throat region, can be recovered by adding only one inch to the combustion chamber length.

Introduction

High pressure, reusable rocket thrust chambers, such as the Space Shuttle Main Engine (SSME), encounter an irreversible plastic deformation and thinning of the cooling passage wall during each thermal cycle. After numerous thermal cycles, cracks form in the cooling passage wall which can lead to failure of the thrust chamber.

A thermal barrier coating (TBC), such as zirconia (ZrO_2), applied to the thrust chamber inner wall could substantially increase the life of the chamber, or allow operation at an even higher chamber pressure. The coating reduces the very high thermal gradient between the hot-gas side wall and the cold back-side closeout. This reduces the cyclic induced plastic strain which causes the deformation of the cooling passage wall.

In recent years, TBCs have been proposed for use on gas turbine blades as well as on rocket thrust chambers. However, their use has not been widely accepted because the coatings are brittle and have a tendency to spall off during repeated thermal cycles. Since the heat flux in a gas turbine is relatively low compared to that in a high pressure rocket thrust chamber, a coating thickness greater than .010 inches is

required to be effective. The throat heat flux in the SSME approaches 100 BTU/in²sec. If a ZrO₂ TBC were to be used on the SSME and the hot-gas side wall temperature allowed to operate at 3500°R, the required coating thickness would be .0005-.001 inches, depending on the apparent thermoconductivity of the coating. Since experience has shown that thin coatings remain better bonded than thick coatings when subjected to thermal cycling, the use of a TBC on a high pressure rocket thrust chamber could be more successful than on a gas turbine, even though the thermal environment in the rocket thrust chamber is much more severe. However, to apply uniform ultra-thin coatings such as this requires some development.

Normally the use of a TBC would be incorporated in the initial design of a thrust chamber. A major drawback in using a TBC for life enhancement on an already-operational rocket thrust chamber such as the SSME is that the coating would reduce the heat which is picked up by the coolant and thus would upset the engine cycle balance. However, this effect could be reduced if only the throat section of the thrust chamber were coated.

The normal procedure of applying TBCs is to plasma spray a bond coat such as NiCr or NiCrAlY onto an already-fabricated thrust chamber and then plasma spray the ceramic layer over the top. Problems associated with this procedure include oxidation of the metallic substrate during the coating application, and a residual stress which remains in the coating when the coated chamber returns to room temperature. Both reduce the bond strength between the coating and the substrate.

Another method of applying the coating which reduces or eliminates the phenomenon described above is to build the thrust chamber inside out, i.e., start with the coating and fabricate the chamber around it. A method of fabricating coated thrust chambers inside out called the "electroform pick-up process," was developed at the Lewis Research Center. The process consists of plasma spraying the ceramic coating onto a mandrel, applying the bond coat, and then electrodepositing a liner around the coating using copper. The cooling channels are then machined into the copper liner after which the channels are closed out by electrodeposition. The mandrel is removed which leaves a smooth ceramic coating which possesses excellent bond strength.

At Lewis there has been an ongoing program to develop techniques of applying ultra-thin TBCs to rocket thrust chambers, both by applying the coating to already-fabricated chambers and by applying the coating using electroform pick-up process. However, the results discussed herein pertain only to thrust chambers having coatings applied by the electroform pick-up process.

In order to experimentally evaluate coatings for use on high pressure rocket thrust chambers, subscale rocket thrust chamber tests were conducted at Lewis using the Lewis plug nozzle thrust chamber test apparatus. Plug nozzle chambers having ZrO₂ coatings .008, .005 and .003 inches thick were fabricated by the electroform pickup process. The chambers were cyclically tested at 600 psia chamber pressure using oxygen-hydrogen as propellants and liquid hydrogen as the coolant. The test results are compared with identical chambers which had no coating.

Also presented are the results of an analysis performed on a high pressure advanced space engine (ASE) design, which shows the effect on the hot-gas side wall temperature, coolant pressure drop and coolant temperature rise if the throat region is coated with a TBC. The analysis also shows the affect of changing the combustion chamber length on the engine cycle balance.

PROGRAM OBJECTIVE

TO DEVELOP THERMAL BARRIER COATING TECHNOLOGY

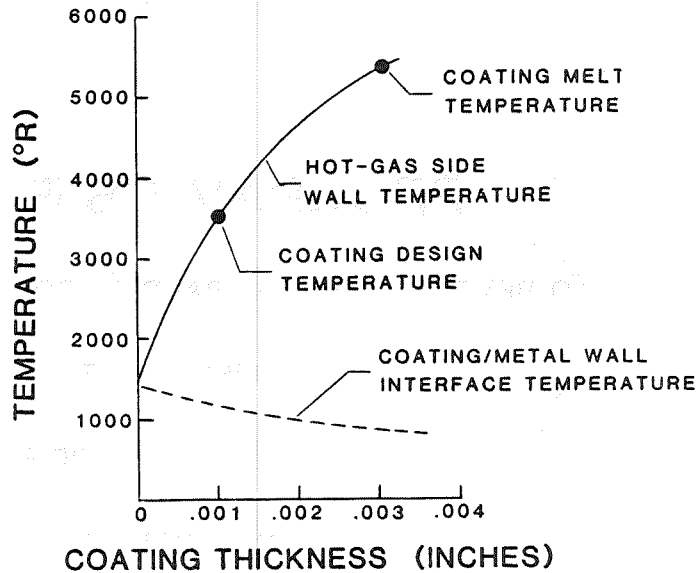
- CHAMBER LIFE EXTENSION
- HIGHER PRESSURE POTENTIAL
- FABRICATION FEASIBILITY
- EXPERIMENTAL VERIFICATION

TECHNOLOGY ISSUES

- COATING DURABILITY
- LIFE EXTENSION, NOT SURVIVAL
- ULTRA THIN COATINGS

THIN COATING REQUIREMENT

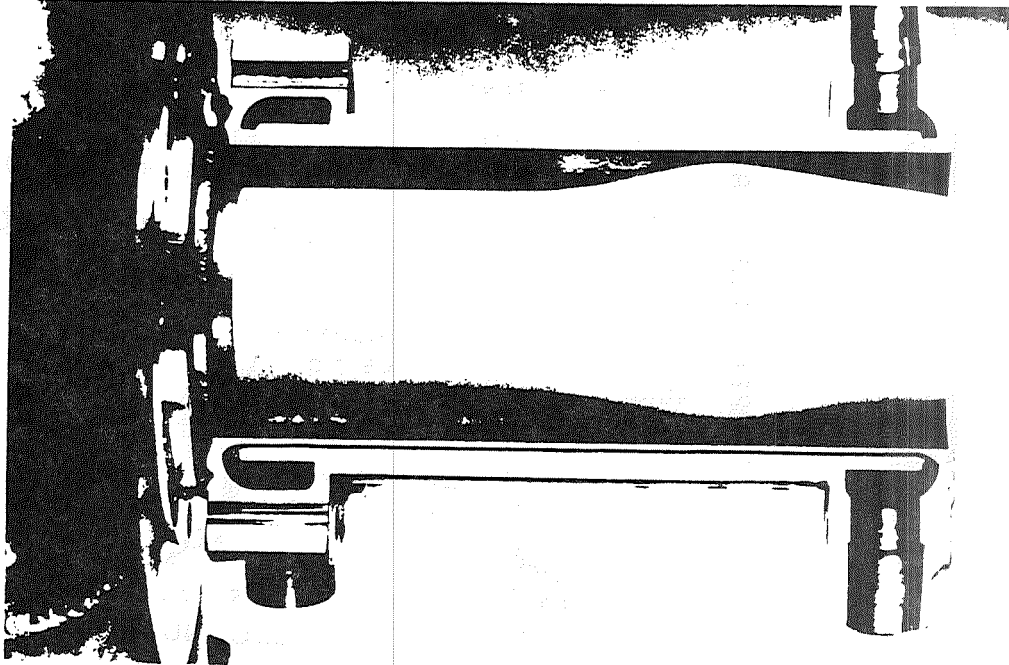
HIGH PRESSURE ROCKET THRUST CHAMBER



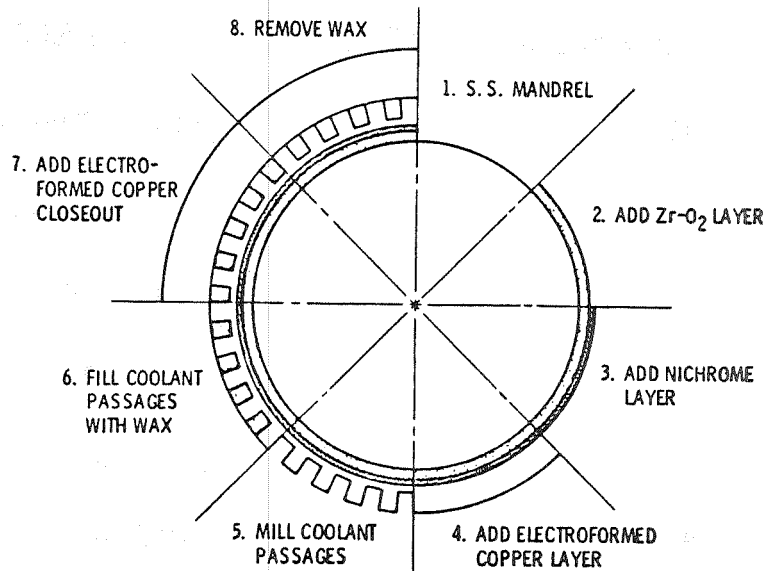
APPROACH

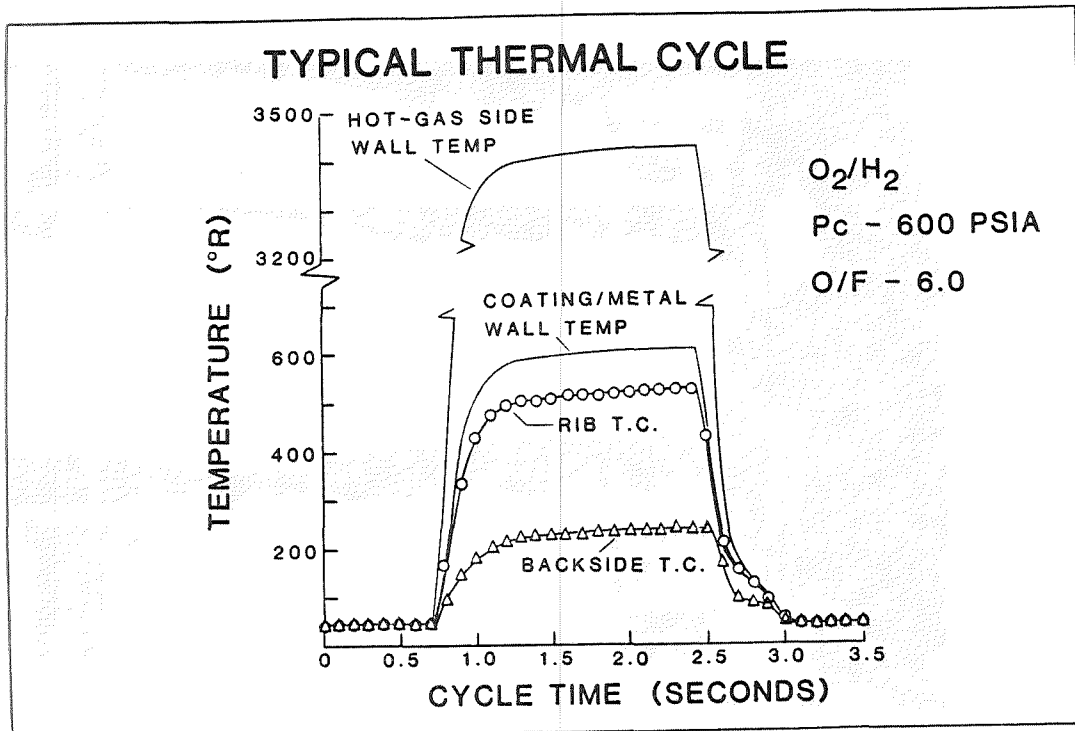
- APPLY COATINGS AS PART OF THE FABRICATION BY THE ELECTROFORM PICK-UP PROCESS
- APPLY COATINGS BY THE ARC PLASMA SPRAY PROCESS TO ALREADY FABRICATED THRUST CHAMBERS

PLUG NOZZLE TEST APPARATUS



CHAMBER FABRICATION SEQUENCE



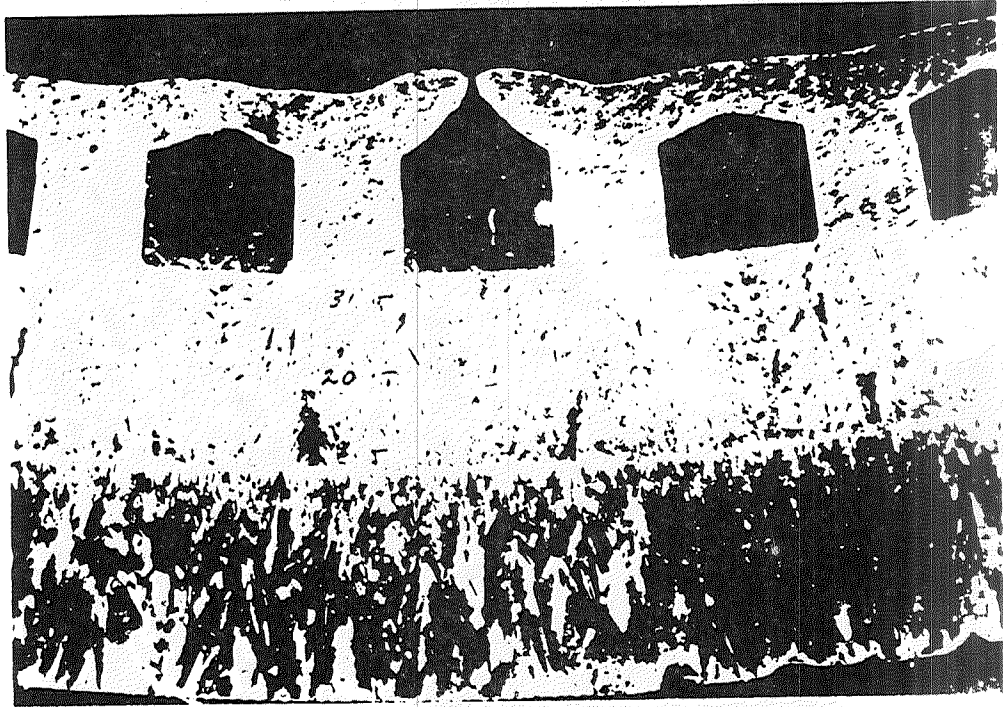


EXPERIMENTAL RESULTS

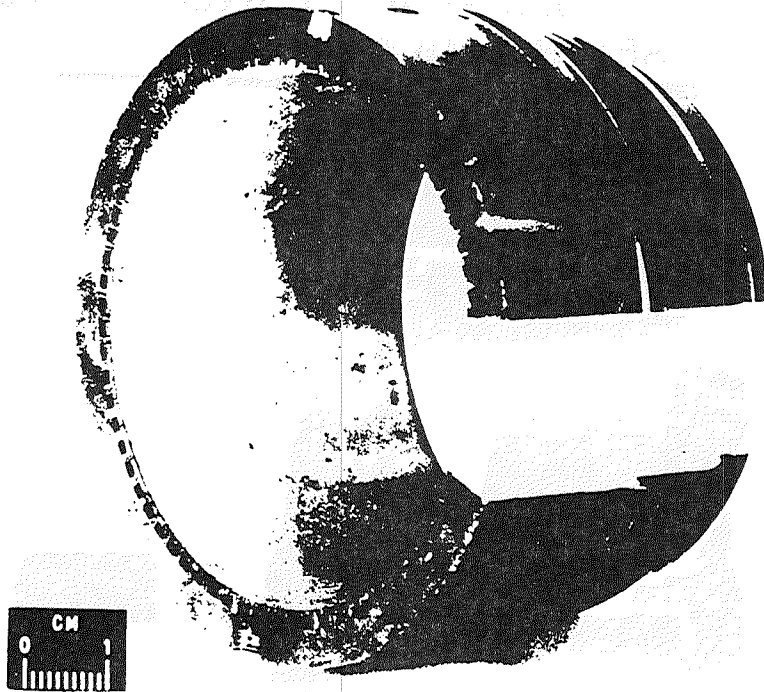
THICKNESS	CYCLES	RESULT
NO COATING	150 AVG	TYPICAL CHANNEL FAILURE
.008"	659	BLOCKED CHANNEL FAILURE
.005"	2413	NO FAILURE
.003"	1538	NO FAILURE
.003"	1448	NO FAILURE

- NO CHANNEL DEFORMATION FOR COATED CHAMBERS
- EXCELLANT METAL TO CERAMIC BOND IN ALL CASES

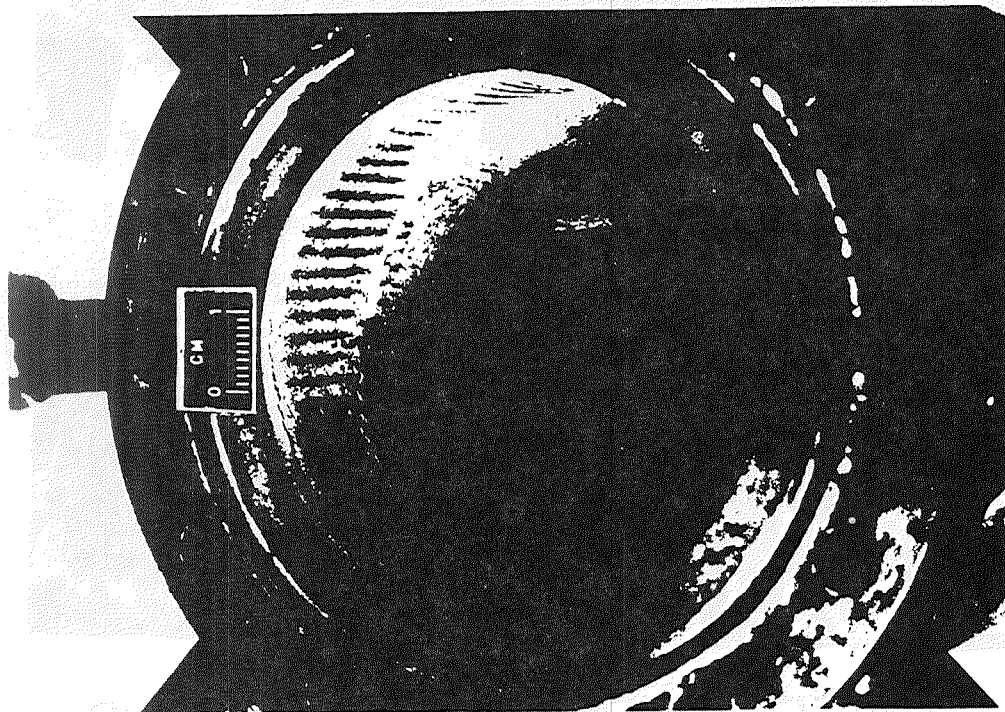
COOLANT PASSAGE FAILURE



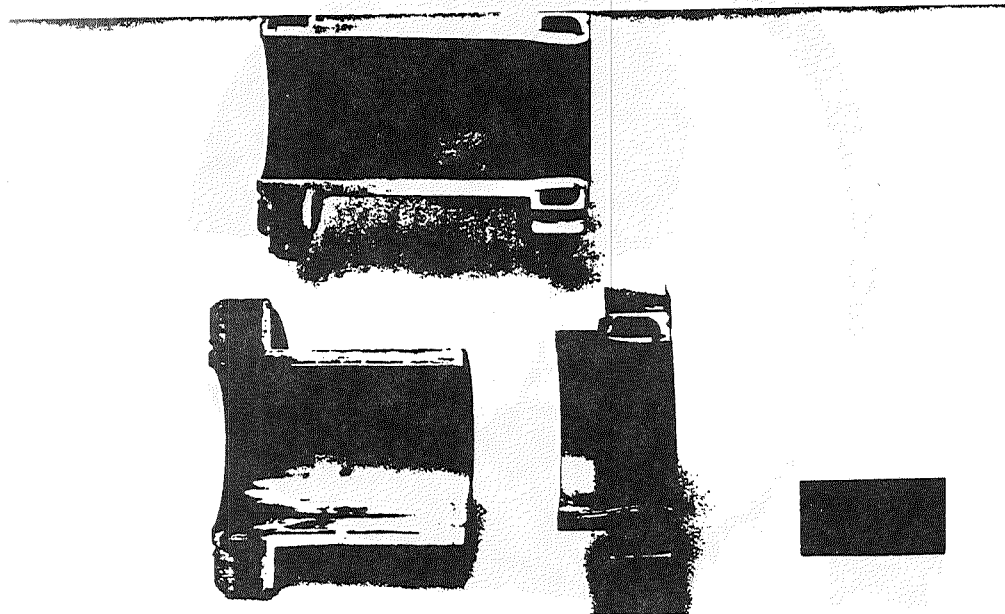
COATED SURFACE CONDITIONS AFTER 659 CYCLES



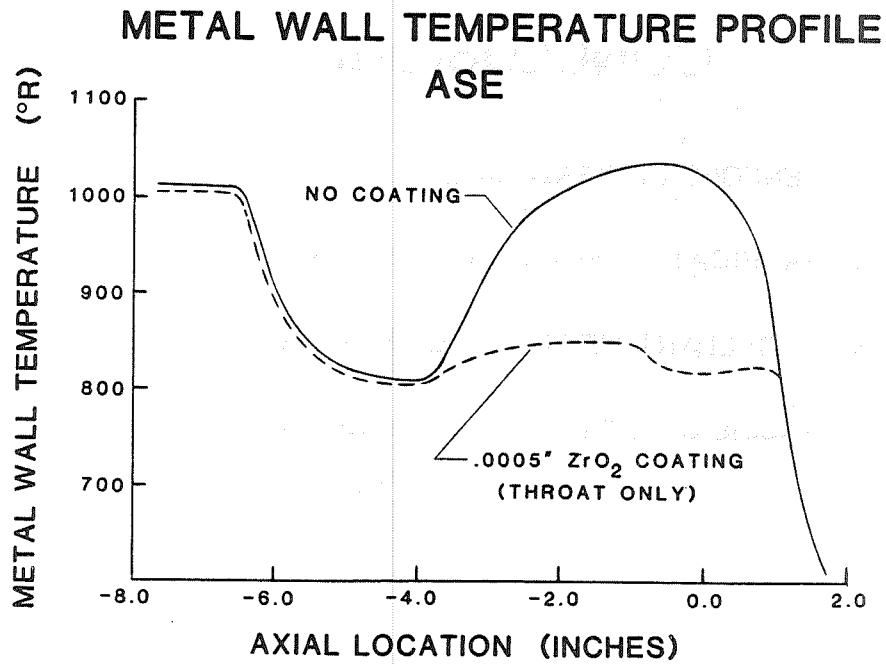
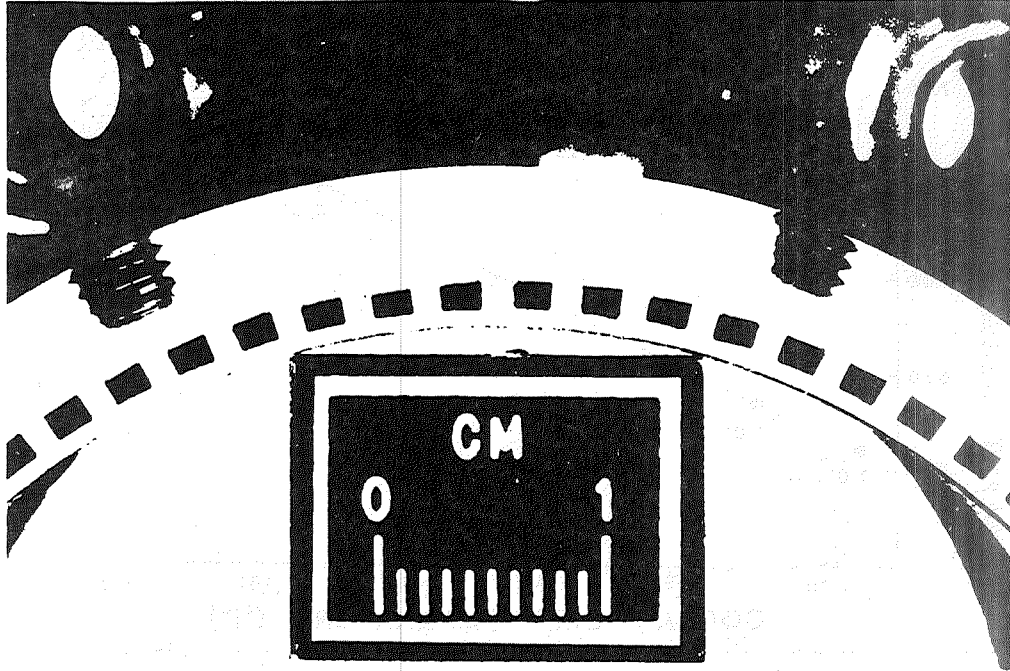
**COATED SURFACE CONDITIONS
AFTER 2413 CYCLES**

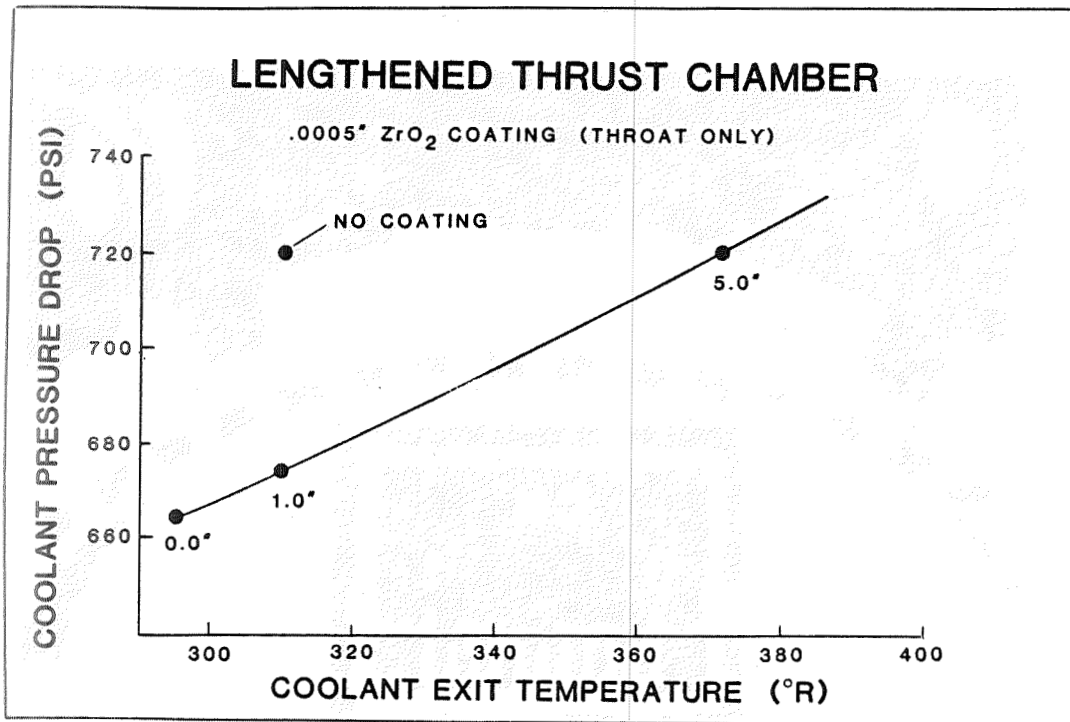


**COATED SURFACE CONDITIONS
AFTER 1448 CYCLES**



COOLANT PASSAGE CONDITION AFTER 1448 CYCLES





CONCLUSIONS

- TENFOLD INCREASE IN LIFE DEMONSTRATED
- FABRICATION FEASIBLE & REPEATABLE
- EXCELLANT METAL TO CERAMIC BOND
- ENGINE CYCLE BALANCE CAN BE MAINTAINED