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Testing of Wrought Iridium/Chemical Vapor Deposition Rhenium Rocket

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SUMMARY

A 22-N class, iridium/rhenium (Ir/Re) rocket chamber, composed of a thick (418-μm) wrought iridium (Ir) liner and a rhenium substrate deposited via chemical vapor deposition, was tested over an extended period on gaseous oxygen/gaseous hydrogen (GO_2/GH_2) propellants. The test conditions were designed to produce species concentrations similar to those expected in an Earth-storable propellant combustion environment. Temperatures attained in testing were significantly higher than those expected with Earth-storable propellants, both because of the inherently higher combustion temperature of GO_2/GH_2 propellants and because the exterior surface of the rocket was not treated with a high-emissivity coating that would be applied to flight-class rockets. Thus the test conditions were thought to represent a more severe case than for typical operational applications. The chamber successfully completed testing (over 11 hr accumulated in 44 firings), and post-test inspections showed little degradation of the Ir liner. The results indicate that use of a thick, wrought Ir liner is a viable alternative to the Ir coatings currently used for Ir/Re rockets.

INTRODUCTION

Fuel film cooling is employed in conventional (silicide-coated C103) radiation-cooled rockets to maintain the material system below the thermal limit of 1370 ˚C. Use of significant amounts of fuel film cooling degrades performance (from incomplete mixing and combustion) and, for some applications, presents a sensor contamination issue (from the unburned fuel in the plume). Rockets composed of a rhenium (Re) substrate and an iridium (Ir) coating have demonstrated long lifetimes at 2200 ˚C operating temperatures. The added thermal margin afforded by iridium-coated rhenium (Ir/Re) rockets allows for the more complete mixing of the film cooling flow into the core flow. As a result, Ir/Re rockets have demonstrated nearly the maximum performance possible with Earth-storable propellants (ref. 1).

The established process for fabricating Ir/Re rockets is chemical vapor deposition (CVD). In the CVD process, a gaseous precursor of the material to be deposited is flowed over a heated mandrel (ref. 2). Upon contact with the mandrel, thermal decomposition of the gas occurs, which results in the deposition of the material onto the mandrel. For Ir/Re chambers, Ir is deposited first over the chamber mandrel (usually 50 to 100 μm thick), followed by deposition of a Re substrate (usually about 2500 μm thick). The mandrel is then etched away chemically; this leaves the free standing chamber.

CVD Ir/Re rockets have been fabricated and tested at 22-, 62-, and 440-N thrust levels (ref. 1). In the initial testing of these rockets, either a cooled section was placed between the chamber and injector or a platinum-rhodium liner was installed in the near-injector region of the chamber (refs. 3 and 4). The added length of the cooled section or liner, both of which employed trip rings, allowed for more complete mixing and the elimination of injector streaking. No significant erosion or degradation of the Ir coating was observed when Ir/Re chambers were tested in this configuration. In more recent testing, Ir/Re chambers have been directly attached to injectors without any intervening cooled sections or liners (the chambers themselves were made longer to allow for complete mixing). In this configuration, degradation of the Ir layer in the form of pitting was often experienced in the near-injector region of the chamber (ref. 5). Apparently, the Ir degradation was a result of exposure of the Ir layer to a mixing, combusting flow field. Several causes have been postulated, including reactivity of the Ir layer with a combustion radical or direct impingement of oxidizers against the chamber wall. A thicker Ir layer may provide enough margin to minimize effect of Ir degradation in the near-injector region of the chamber. The CVD process, however, would not be a practical way of fabricating thick Ir layers. Machining from wrought materials provides a more reliable and cost effective way of fabricating a thick Ir liner. For this study, a three-piece, 418-μm-thick Ir liner was fabricated from wrought materials. The Re substrate was applied over the wrought Ir liner by using CVD. The rocket was tested on gaseous oxygen/gaseous hydrogen (GO_2/GH_2) propellants at a mixture ratio that simulated Earth-storable combustion

environments. The testing was conducted to demonstrate the integrity of the wrought Ir liner in a rocket combustion environment.

TEST APPARATUS

Rocket Chamber Geometry

The rocket chamber geometry used for this study had a standard design used by NASA for screening advanced chamber materials (refs. 6 and 7). The barrel had a nominal inner diameter of 0.918 cm, while the throat inner diameter was 0.431 cm. The contraction ratio was 4.5:1 with a gradual converging contour to the throat. The nozzle was an 8:1 area ratio, 15° half-angle cone. A 37° angle cone at the head end was used for attachment to the injector assembly. The rocket was sized for mass flowrates corresponding to a 22-N thrust level.

Rocket Chamber Fabrication

The wrought Ir liner was fabricated by Tecomet and consisted of the attachment cone, the barrel section, and the throat section, as shown in figure 1. The barrel section was rolled from Ir sheet stock into a tubular piece and electron beam (EB) welded longitudinally. The attachment cone and the throat section were electron discharge machined from solid Ir bar stock. The attachment cone and throat section were circumferentially EB welded to the barrel to form the Ir liner. All of the EB welds had 100-percent penetration and were demonstrated to be leak tight at 207 kPa for 30 min. The Ir liner had a nominal thickness of 418 μm. The finished Ir liner is shown in figure 2. The Re substrate was deposited over the Ir liner by using CVD (by Ultramet) and had a nominal thickness of 0.11 cm.

Injector Assembly

The chamber was attached to an assembly consisting of a water-cooled transition section and injector. The injector body had a center annulus containing radial injector ports and a spark plug. Oxygen was injected radially into the center annulus to flow around the spark plug, which was used to energize the oxygen flow. A small amount of hydrogen was injected radially downstream of the spark plug to ignite the energized oxygen. Further downstream, six impinging elements injected hydrogen into the center annulus flow for main ignition, while six elements injected hydrogen axially for fuel film cooling of the near-injector region.

The 5.08-cm-long, water-cooled section had a trip ring 2.54 cm downstream of the injector face to promote both, mixing of the fuel film into the core flow and the elimination of streaking. The water-cooled section also protected the front end from thermal soakback from the chamber. The conical head end of the chamber was clamped to a cone on the face of the water-cooled section by a split ring and sealed with a flexible graphite gasket material.

Purge Assembly

The exterior surface of the rocket chamber was surrounded by argon to prevent the oxidation of the exterior Re. A 24-element ring was used for injection of the purge gas around the chamber exterior. The purge was contained in a 5.20-cm-diameter fused silica tube, with a 2.54-cm inner diameter, stainless steel end cap. An oxygen absorbing purifier was used in the purge line to ensure that the purge gas had less than 0.1 ppm of oxygen. The exterior purge was initiated prior to the first test in each series and maintained until after each series was completed and ambient pressure had been reestablished in the test tank (by which time the chamber had cooled to ambient temperature).

Instrumentation

A pressure tap in the injector face was used to measure the static chamber pressure. This static pressure measurement was corrected for momentum pressure loss across the combustion zone and converted to total pressure. Hydrogen and oxygen mass flowrates were calculated using the inlet pressures, inlet temperatures,

and discharge coefficients of critical flow venturis, with corrections for real gas effects. The measurement uncertainties were calculated to be less than ±2 percent for all of the mass flowrates and mixture ratios.

Outer wall chamber temperatures were measured using a two-color pyrometer. The pyrometer was aimed at the converging section on the chamber (typically the highest temperature) and had an approximately 1.0-cm-diameter spot size. The temperature range of the pyrometer was 982 to 3315 ˚C, with a measurement uncertainty of approximately ±30 ˚C. Inner wall temperatures were estimated from the outer wall temperature measurements and an energy balance between the one-dimensional, steady-state heat conduction through the chamber wall and the radiation emitted from the chamber outer wall. The methodology for this temperature estimate is explained in detail in reference 6.

Test Facility

Testing of the chambers was conducted in a propulsion test facility designed for low-thrust rockets operating on GO_2/GH_2 propellants. The rocket was mounted in a 0.91-m cylindrical test tank with viewing ports for optical access. A two-stage air ejector system maintained a 1.4-kPa pressure in the tank, equivalent to an altitude of 36.6 km. The rocket was mounted horizontally and fired into a water-cooled diffuser. All data were recorded on a PC-based data acquisition system and performance parameters were calculated in real time. A more detailed description of the test facility is available in reference 8.

TEST RESULTS

A total of 44 firings and 40 347 sec (11.21 hr) were accumulated on the wrought Ir-lined chamber without failure. The majority of the tests were 1200 sec in duration. The nominal test condition was 510-kPa chamber pressure and a mixture ratio of 4. If complete combustion was assumed, this test case provided conditions comparable to the oxygen (0.23-kPa partial pressure), atomic oxygen (0.41-kPa partial pressure), hydroxyl (8.6-kPa partial pressure), and water vapor (248-kPa partial pressure) content in an Earth-storable propellant combustion environment at 690-kPa chamber pressure. Since GO_2/GH_2 combustion temperatures are higher than those obtained with Earth-storable propellants, the test condition was considered to be a more severe environment. The GO_2/GH_2 testing, of course, could not simulate the liquid injection phenomenon (droplet atomization and vaporization) nor the nitrogen oxide content present in Earth-storable propellant combustion. The total test time accumulated is more than would be required for qualification of a flight engine (for most applications).

Visual inspection of the EB welds in the Ir liner showed no signs of degradation and the Re substrate appeared to be adherent to the Ir liner. Dimensional checks showed that there was no change in the throat diameter during testing. Early in the testing, the throat region interior took on a crystalline appearance, evidence of grain growth from high temperatures experienced in the throat region. The outer wall temperature varied from 2100 to 2300 ˚C, with estimated inner wall temperatures ranging up to 2325 ˚C. The inner wall at the throat region, then, was within 115 ˚C of the Ir melting point (2440 ˚C) during some points of the testing. In a flight chamber, a high emissivity coating would be applied to the exterior to provide better heat rejection from the chamber and thus avoid grain growth in the throat.

There was a small amount of the Re layer removed from the end of the nozzle. This was likely a consequence of nozzle exit extending out the end of the purge tube. The high-temperature plume from the underexpanded nozzle undoubtedly oxidized the portion of the Re substrate not protected by the exterior purge. The total mass loss from the chamber in this testing was on the order of 0.5 percent, the majority of which came from the nozzle end. Grain growth was also seen on the Re exterior upstream of the throat. This was likely from an incident at the end of the second test, when the exterior purge was compromised by a washback of the plume.

Table I shows the test history of the wrought Ir-lined chamber and along with those for a CVD Ir/CVD Re chamber and a chamber composed of a powder metallurgy (PM) substrate and electroformed deposited (ED) coating (ref. 7). All of the chambers were of the same geometry and were tested with similar injector apparatus on GO_2/GH_2 propellants. CVD is the established process for fabricating Ir/Re rockets, while the ED Ir/PM Re process is the most matured alternative fabrication method. The wrought Ir-lined chamber accumulated test time (>11 hr) comparable to the Re chambers with CVD and ED Ir coatings. The wrought Ir-lined chamber, however, was tested at higher temperatures than the ED Ir-coated chamber (there were no reliable temperature measurements of the CVD Ir-coated chamber). The testing in this study, then, demonstrates the integrity of the wrought Ir liner in a rocket combustion environment and shows that it is a viable alternative to CVD and ED Ir coatings for Re chambers.

CONCLUDING REMARKS

A wrought iridium (Ir)/chemical vapor deposited (CVD) rhenium (Re) chamber was tested for 11.21 hr and 44 firings on gaseous oxygen/gaseous hydrogen (GO_2/GH_2) propellants without failure. The test condition simulated the oxygen, atomic oxygen, hydroxyl, and water vapor content of an Earth-storable propellant combustion environment, but at a higher temperature. The Ir liner was composed of three pieces (the attachment cone, barrel section, and throat section) fabricated from wrought material and electron beam (EB) welded together. A thick Ir liner may be a way to minimize the effects of an Ir degradation phenomenon seen in the near-injector region of some Ir/Re chambers. There was no dimensional change in the throat diameter and no evidence of degradation of the EB welds in the Ir liner. The testing demonstrates the integrity of wrought Ir liner in a rocket combustion environment. The time accumulated on this chamber compares favorably with that of CVD Ir/CVD Re and ED Ir/powder metallurgy (PM) Re chambers tested under similar conditions.

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TABLE I. —COMPARISON of 22-N CLASS, Ir/Re CHAMBERS TESTED ON GO_2/GH_2 PROPELLANTS

Fabrication process	Chamber pressure, kPa	Mixture ratio	Outer wall temperature, ℃	Firings	Time, hr
CVD Ir/CVD Re[a]	682 424 486	3 to 4 5 to 6 7 to 8	(b) (b) (b)	78 35 9 Total: 128	11.33 2.85 0.03 Total: 14.21
ED Ir/PM Re	496 496	3.2 4.2	1700 to 1800 1900	84 7 Total: 91	9.04 2.41 Total: 11.45
Wrought Ir/CVD Re	510	4.1	2110 to 2300	44	11.21

[a]Testing was stopped because of a facility-related failure.

[b]No reliable temperature measurement was made.

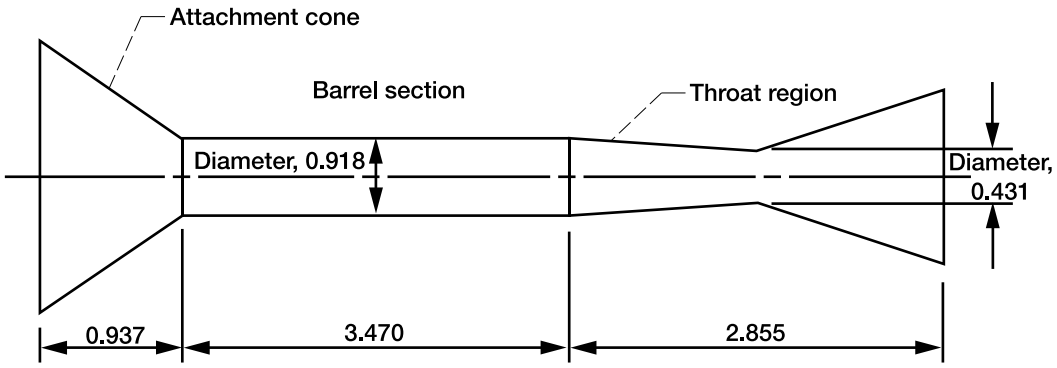

Figure 1.—Iridium liner drawing (all dimensions in centimeters).

Figure 2.—Finished iridium liner.

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