THERMAL BARRIER COATING FOR LIQUID ROCKET ENGINE THRUST CHAMBER.

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Abstract. *Due to propellants combustion reactions, high temperatures acts in the walls of Liquid Rocket Engine (LRE) thrust chamber. This phenomenon can compromise the structural integrity of the material used to manufacture the thrust chamber. One way to decrease this high temperatures is apply a Thermal Barrier Coating (TBC) over the inner surface of chamber walls. The purpose of this paper is show the thermal behavior of 8wt % yttria stabilized zirconia TBC apply over copper alloys for aplications in LRE thrust chamber. There are possibilities to apply TBCs over the internal surface of LREs walls thrust chambers to attend the Brazilian Space program. A holder was prepared to analyze the heat flux through coated and uncoated with 0.1 mm thick TBC applied over copper alloy (UNS C18200) samples. Before introduction in a plug flow combustion chamber with a large ratio length – diameter and a premixed combustion system to simulate the real work conditions and a premixed combustion system to simulate the real work conditions, the samples were fixed in the holder, negelecting the heat losses from the sides. With this configuration, it's possible measure the temperature 0.1 mm over the surface samples in contact with hot gases using a thermocouple. Preliminary results from the test stand allow check the temperature versus combustion equivalence ratio. Experimetal results from the heating and coolig samples are showed to discuss the TBC thermal behavior. From this results will be possible estimate the thermal characteristics of the TBC analyzed.*

Keywords*: thermal gradients, thermal barrier coating, zirconia*

1. INTRODUCTION

In 1997, Immich and Mayer published a paper show new technologies that would be used in the LRE's design and manufacture by European Space Agency (ESA). Among these technologies were the application of thermal barrier coatings and increase the wall thrust chamber thickness. The National Aeronautics and Space Administration (NASA) have developing technologies for a new advanced launch vehicles generation with efforts to increase the reliability of Reusable Launch Vehicle (RLV) (Raj et al, 2007).

For over 20 years, Thermal Barrier Coatings (TBCs) have been in use in aero turbine engine hot sections. The initial applications were driven by the need to suppress component degradation caused by excessive thermal gradients in vane airfoils (Gleeson, 2006). This analysis about work conditions can be apply for LRE thrust chamber, because it works in a high level of environment temperatures, situation that can influence the thrust chamber structural integrity. Aiming to understand the thermal behavior of a TBC system there are some researches lines (see Greuel et al, 2002 and Winterfeldt et al, 2005).

The temperature in the supersonic nozzle region can achieve range of 1200 K and 2400 K, so it's necessary to use significantly wall thickness with the purpose to dissipate the heat from the combustion reactions. The function of nozzles is to expand the gases in conditions of high temperature and high pressure up to external environment pressure, in order to give impulse to the rocket.

Östergren and Wirgren in a paper published in 2007 discussed how the company Volvo Aero Corporation (VAC) has introduced the system coatings in the form of thermal barrier in the thrust chamber of Viking 4 engine, as well in versions 5 and 6, successor of Viking 4, Ariane 4 rocket propulsion. Until the retirement of this engine in 2003, over of 1000 chambers were coated and launched in more than 130 rockets.

TBC consist of a bonding layer and a top layer. The bonding layer is a transition layer between the top layer and the base material. Its function is to effectuate chemical adhesion of the top layer, to provide corrosion resistance of the base material and to compensate the different thermal coefficients between the top layer and metallic substrate material. The top layer is the actual thermal barrier layer (Greuel et al, 2002) because part of this system is manufactured with materials that have low thermal conductivity, like ceramics.

Although a TBC system is manufactured with low thermal conductivity materials, its efficient operation depend from the existence of thermal gradient between coated metallic substrate surface in contact with hot gases from the combustion reactions and the opposite metallic substrate surface. An alternative to maintain the thermal gradient necessary for the efficient TBC's work is the use of a regenerative cycle of cooling, which is used to introduce the fuel in the thrust chamber. The other hypothesis that it's necessary to be considered is the steady state heat transfer.

In this moment, IAE/DCTA is studying the possibility to apply TBCs in internal surface of wall's thrust chambers of LREs. A LRE named L75, propelled by turbo pumps system is in development. This LRE use a regenerative cooling system so would be good possibility to apply and check the efficiency of TBC apply over the inner wall thrust chamber.

2. HEAT FLUX THROUGH TBC SYSTEM

Different computer programs, as well as different methods of calculation have been used as the objective of calculating the heat flux thorough the wall thrust chamber of LRE. Some methodologies are based on the theory of divide the wall thrust chamber by finite elements. The heat flux calculations are generally based on the thrust chamber structural design.

Following the methodology proposed by Kesaev, 1997, cited by Almeida, 1999, it's possible to estimate heat flux acting on the wall thrust chamber.

The initial parameters for calculating the heat flux through the wall thrust chamber are:

- Thrust chamber contour (see blue contour in Figure 1);
- Pressure acting in thrust chamber;
- Reactions combustion equivalence ratio;
- Temperature acting in the inner wall thrust chamber surface;

It's considered a simplification in the heat transfer calculations: the inner wall temperature is in the 800K to 1000K range.

The first step is to calculate the thermo-physical parameters of the combustion reactions, known the initial parameters of mixture ratio, pressure and combustion temperature.

In the second calculation step the thrust chamber is divided in 10 to 15 parts and for each of these divisions apply the basic correlations presented in Eq. (1), Eq. (2) and Eq. (3) in relation to convection and radioactive heat flux.

$$q_{co} = \left(B \frac{1-\beta^2}{{D_{sp}}^{1,82}} \right) \cdot \left(\frac{{P_{ch}}^{0,85}}{{D_{cr}}^{0,15}} \right) \cdot \left(\frac{So}{\mathrm{Pr}^{0,58}} \right) \tag{1}$$

$$B \approx 0.0086 \tag{2}$$

$$\beta = \sqrt{1 - \frac{2}{2 + M^2 (k-1)}} \tag{3}$$

$$D_{sp} = \frac{D}{D_{cr}} \tag{4}$$

Where:
B is a non dimensional coefficient;
β is a non dimensional velocity;
S_o is a complex thermophysival parameter dependent on the mixture ratio and wall temperature (range: 3000 to 7000 W/m^2) obtained from experimental tests;
Pr is the Prandtl number (for gases in the wall layer condition Pr ~ 0.75);
M is the Mach number (range: 0 a 4.5);
k is the adiabatic exponent (range: 1.14 to 1.21);
P_{ch} is the chamber pressure (MPa);
D is the actual section diameter (m);
D_{cr} is the critical diameter (m);

The basic correlation for calculation of the radioactive heat flux

$$q_{rad} = \varepsilon_{wall} \varepsilon_g \sigma ({T_{ch}}^4) \varphi \tag{5}$$

Where:
σ is the Steffan-Boltzmann coefficient (5.67 . 10^{-8} W/m^{2}K^4);
T_{ch} is the gas temperature at core of combustion chamber (K);
ε_{wall} is the effective emissivity coefficient of wall surface (less than 1);
ε_g is the emissivity coefficient of chamber gases (less than 1);

φ is the absorption coefficient in wall layer (less than 1).

The heat flux calculated from this methodology is presented in Figure 1.

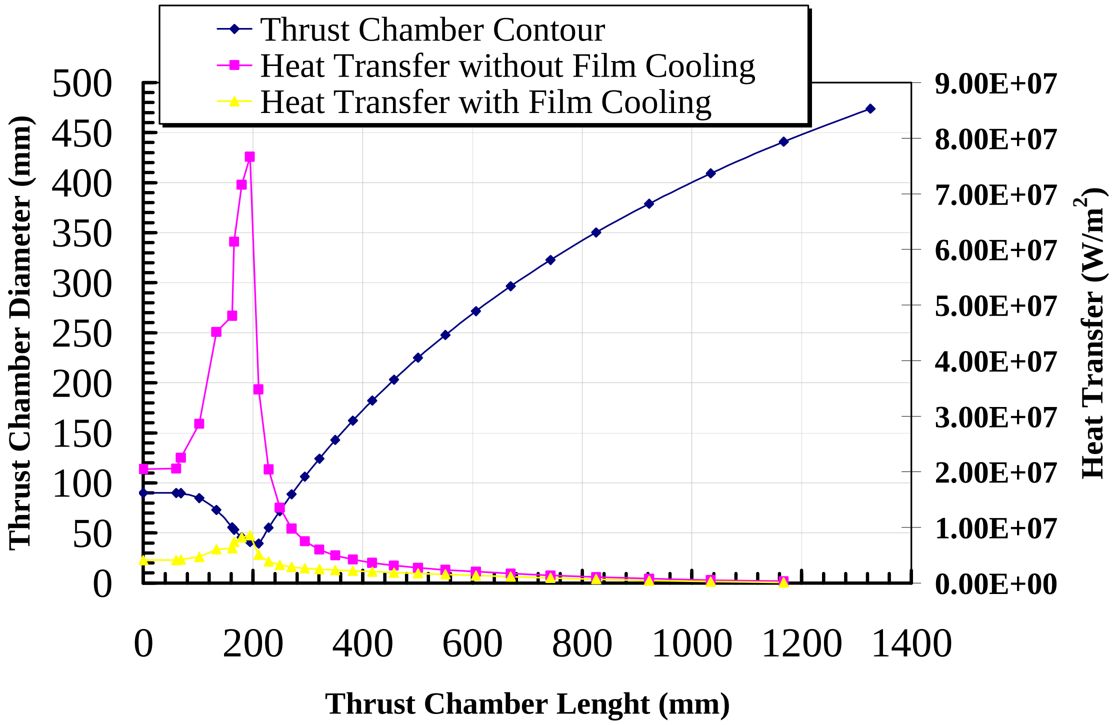

Figure 1. Heat flux through the wall thrust chamber.

In Figure 1 the pink line shows the heat flux through wall thrust chamber without film cooling and yellow line shows the heat flux through with the film cooling action. The difference from this two configurations is fairly representative, because is explicit that the film cooling action decrease significantly the heat flux action through the wall thrust chamber.

The film cooling happen when there is an intentionally fuel injection by pattern injectors that result in chamber walls cooling. Other important analyze is that the high performance injectors can increase the heat transfer through the combustion chamber walls and nozzle walls (Sutton 2001).

The heat flux calculation allows knowledge of the temperatures acting on the wall chamber, from this results Greuel et al., 2002, proposed the following analysis: taking T_w like the temperature in the interface between the coating and metallic substrate and T_C like the temperature in the inner wall surface in contact with hot gases, it's possible to calculate the heat flow through the thrust chamber wall in the steady heat transfer expressed by Equation 6:

$$q_{conv} = \alpha_c.(T_c - T_w) \tag{6}$$

Where q_{conv} is the heat convective flux acting on the wall thrust chamber and α_c is the heat transfer coefficient of the hot gases.

The heat transfer part by conduction can be estimated by Equation 7:

$$q_{cond} = -\lambda.\frac{\Delta T}{\Delta y} \tag{7}$$

Where q_{cond} is the heat convective flux through the wall thrust chamber, λ is conductivity coefficient of ceramic material; ΔT is the temperature difference between the opposite surfaces of the coating and Δy is the coating thickness.

Neglecting the radioactive heat transfer it's possible to write the following energy balance by Equation 8:

$$q_{conv} = q_{cond} \tag{8}$$

The energy balance presented allows calculate the temperature acting in the interface between the metallic substrate and ceramic coating. This temperature can be obtained by Equation 9, which is obtained from the algebric manipulation of energy balance shown in Equation 8:

$$T_w = T_c - \frac{\Delta y}{\lambda} Q \tag{9}$$

Where Q is the wall heat flux.

A comparison between TBCs applied over inner surface of LRE thrust chambers and turbine blades is presented in Table 1 showing the difference temperatures obtained for these different applications.

Table 1: Comparison of characteristics data for liquid rocket engines and turbines blades (Greuel et al., 2002)

	Inner Wall Thrust Chamber Surface	Turbine Blades
Q (MW/m2)	82,5	1,5
Δy (μm)	10	150
λ (W/mK)	1,5	1,5
ΔT (K)	550	150

3. EXPERIMENTAL TEST STAND

The material temperature increase and material expands as the energy from the heat is absorbed. The temperature gradients allows that the energy be transported from hot regions of the material to cooler regions (Callister, 2008).

The thermal characterization for coated and uncoated copper alloy samples assumes the steady state heat transfer hypothesis, in other words the heat flux that goes through any of the samples perpendicular planes doesn't change with the time.

The aim of this experimental apparatus is analyze the heat transfer that happening through the TBC system and make an experimental studies to check if the high thermal gradients is decreasing because of TBC insulating.

The study of copper alloy samples thermal behavior is given by thermal cycles of heating and subsequent cooling. The steady state heat transfer temperature is measured by thermocouples in the coating and metallic substrate interface.

One of the difficulties of this experimental work is how to insert the thermocouple close the surface in contact with the hot gases. The solution was made a hole in the copper sample up to 0.1 mm over the TBC (see Figure 2). A holder was manufactured to fix the sample over the combustion process, see Figure 3, and analyze the heat flux through uncoated and coated samples. Other particularity from this experimental work is maintaining the thermal gradient between coated surface and it's opposite surface in metallic substrate. In this case the sample was positioned over the plug flow combustion chamber opened to atmosphere, see in Figure 2.

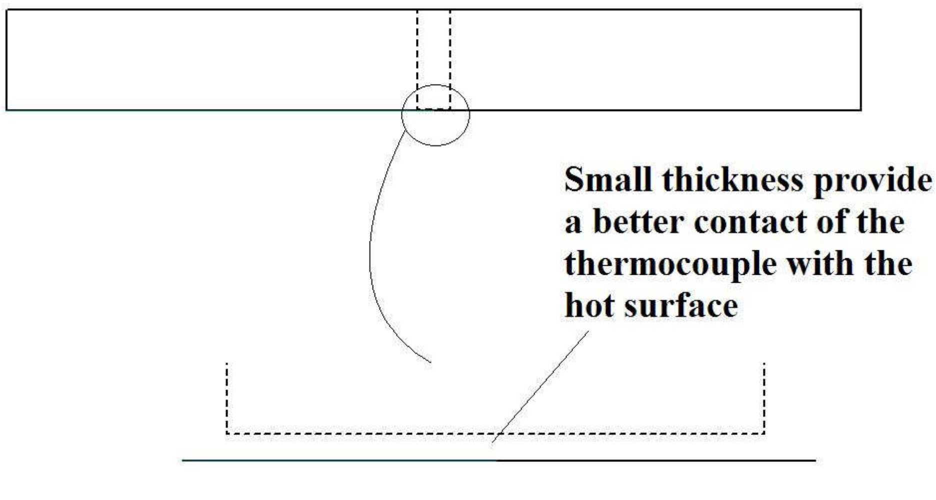

Figure 2. Hole for the insertion of the thermocouple.

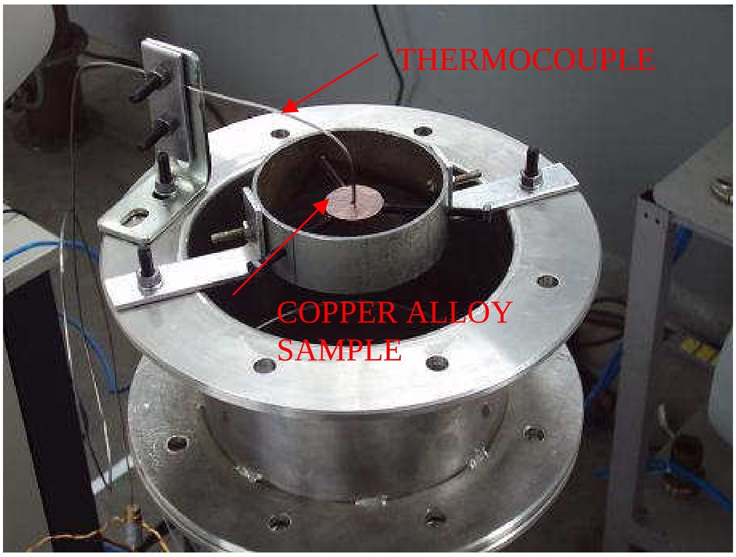

Figure 3. Samples holder.

This device is positioned over a plug flow reaction chamber, which is powered by methane and air propellants, using a premixed combustion system. This combustion chamber was developed from the plug flow model, showed in Figure 3.

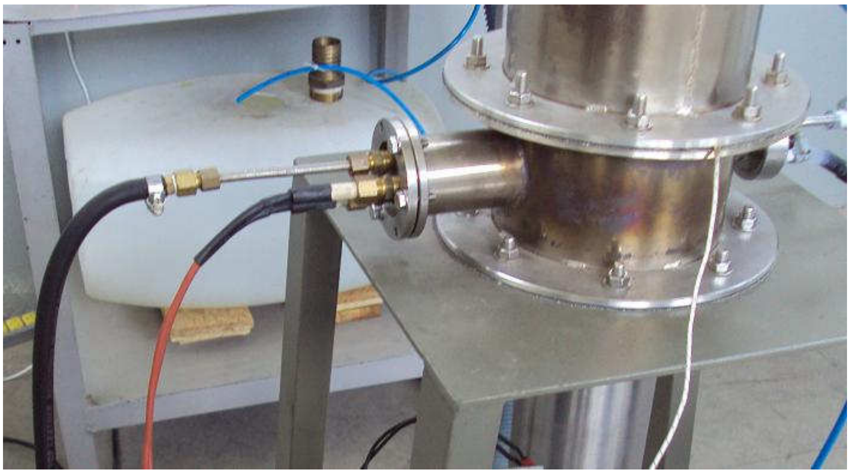

Figure 4. Plug flow reaction chamber.

The plug flow model provides to a sample in analyze a one dimensional heat transfer, the same heat transfer situation presented by Thermal Barrier Coating system. According Turns, 2006, a plug-flow reactor represents an ideal reactor that has the following attributes:

1. Steady – state, steady flow.
2. No mixing in the axial direction. This implies that molecular and/or turbulent mass diffusion is negligible in the flow reaction.
3. Uniform properties in the direction perpendicular to the flow, one dimensional flow. In any cross section, the flow velocity, temperature, composition characterizes the flow completely.
4. Ideal frictionless flow. This assumption allows the use of the simple Euler equation to relate pressure and velocity.
5. Ideal-gas behavior. This assumption allows simple state relations to be employed to relate T, P, ρ, and h.

The combustion chamber showed in Figure 4 has a premixed combustion system where the propellants are in mixed condition before the combustion reactions. According Mahallawy and Habick, 2002, a flame may be both stationary flames on a burner and propagating flames by a flow of gas from a burner tube, or may be freely propagating flames traveling in an initially quiescent gas mixture. Stationary flames are of two general types:

- Premixed flames where the reactants are mixed before approaching the flame region. These flames can only be obtained if the initial fuel and oxidant mixture lies between certain composition limits called the composition limits of flammability.
- Diffusion flames where both mixing of fuel and air and the combustion will happen at the interface.

In a work situation the wall thrust chamber will be submitted to a premixed combustion chamber. This test stand was manufactured thinking in observe the samples in real work conditions. The Figure 5 shows the experimental test stand that was assembled to obtain the experimental results, where were used a plug flow combustion chamber, a thermocouple type K and a computer to acquisition of the experimental data.

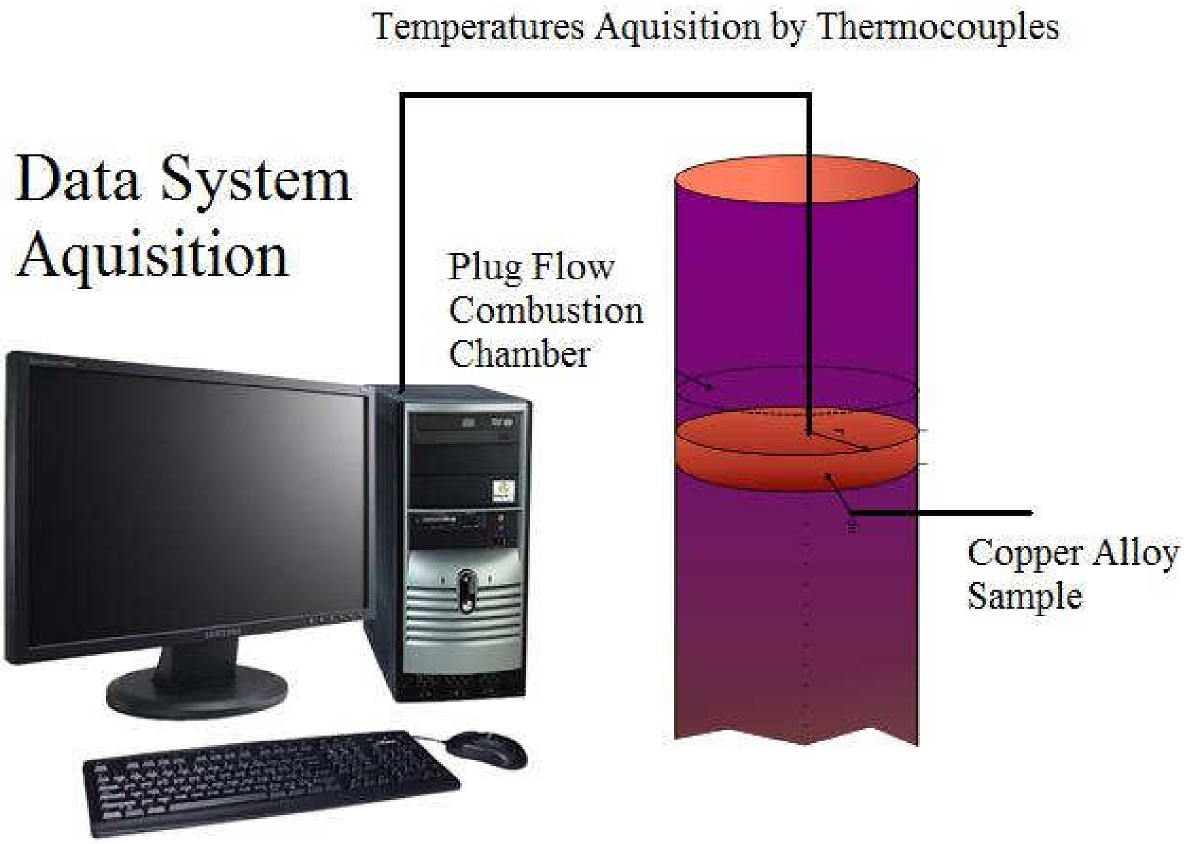

Figure 5. Experimental test stand scheme.

LabWiew was the data acquisition software used to collect the datas from the experimental work. This is commercial software developed by the National Instruments, among it's advantages is the possibility to connect the thermocouples to data acquisition boards.

It's expected that the temperature in the surface in contact with hot gases with ceramic thermal coating will decrease when compared with the same surface without ceramic thermal coating.

4. RESULTS

The temperature acting on the sample depends of the combustion equivalence ratio. According Turns, 2006, the combustion equivalence ratio is commonly used to indicate quantitatively whether a fuel-oxidizer mixture is rich, lean, or stoichometric. The stoichometric quantity of oxidizer is just that amount needed to completely burn a quantity of fuel. If more than a stoichometric quantity of oxidizer is supplied, the mixture is said to be fuel lean, or just lean, while supplying less than stoichometric oxidizer results in a fuel-rich, or rich mixture.

Some experiments were made adding fuel to mixture (methane and air) and measuring the temperature. These preliminary tests allows the knowledge that the temperatures that would be achieved according the combustion equivalence ratio of propellants.

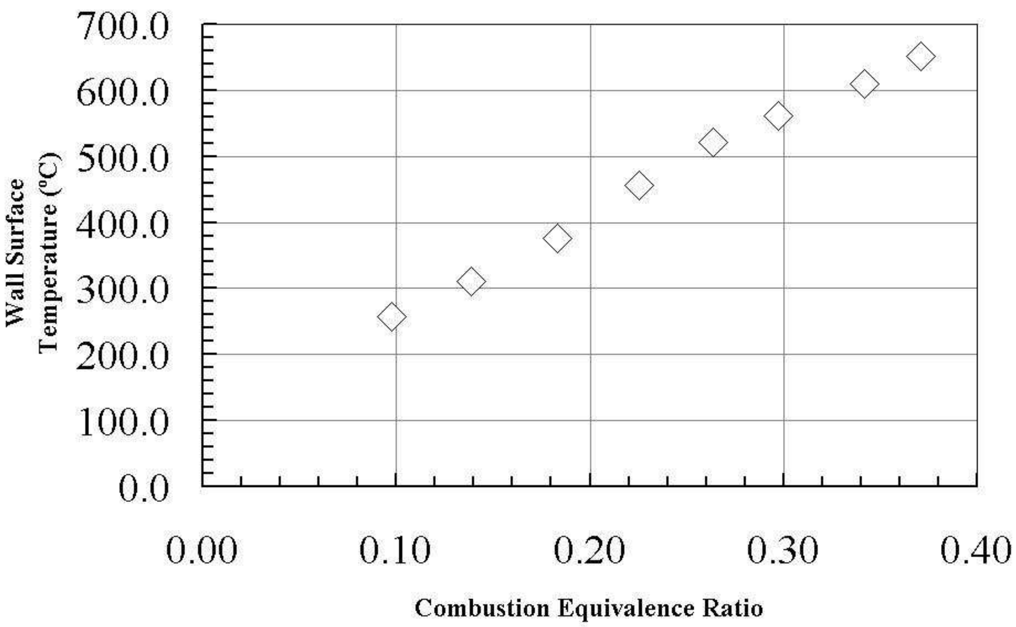

Figure 5. Wall surface temperature versus combustion equivalence ratio.

The Figure 5 shows the temperature behavior accord to the combustion equivalence ratio. From these results it's possible to control the equivalence ratio and submit the coated sample at interest temperatures to determine in qualitatively form the coating efficiency. These temperatures are the same that adiabatic flame temperature of the plug flow chamber used to heat the samples.

The combustion temperature depend on the reactants initial conditions (pressure and temperature), reactants compositions, the process type involved (constant pressure or constant volume) and principally equivalence ratio.

It's possible define two adiabatic flame temperatures: one for constant pressure combustion and one for constant volume. The combustion chamber used in this experimental work was modeled based on constant pressure model (Turns, 2006).

Samples with and without coating were submitted to combustion processes with 0.1 and 0.3 combustion equivalence ratio until to get temperatures of steady state heat transfer, 260 ^{0}C (530 K) and 600 ^{0}C (870 K) respectively. The Figures 6 and 7 present the thermal behavior for the samples analyzed.

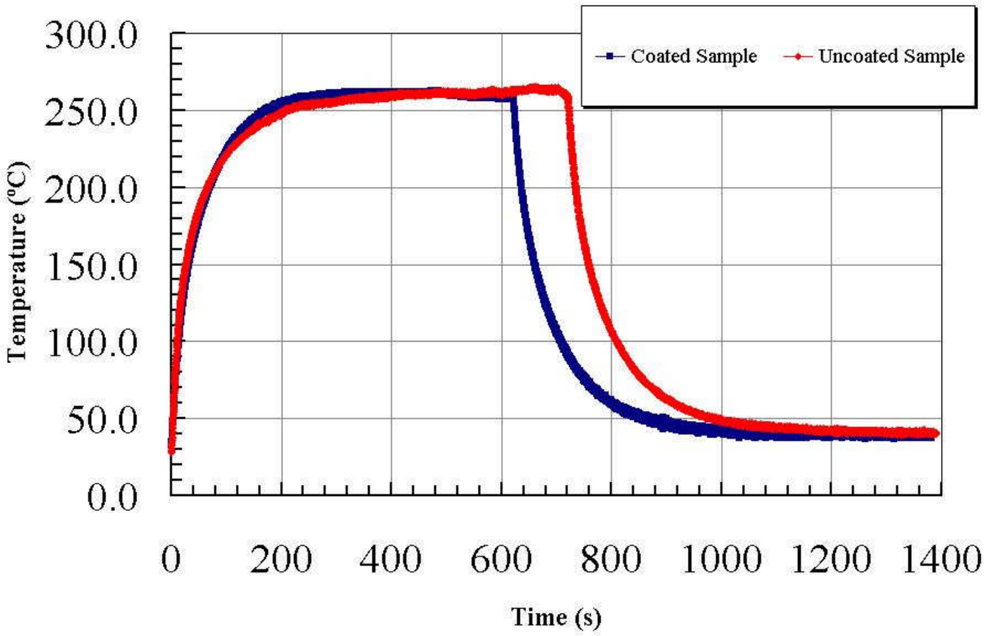

Figure 6. Heating and cooling curve for coated and uncoated sample for 0.1 combustion equivalence ratio.

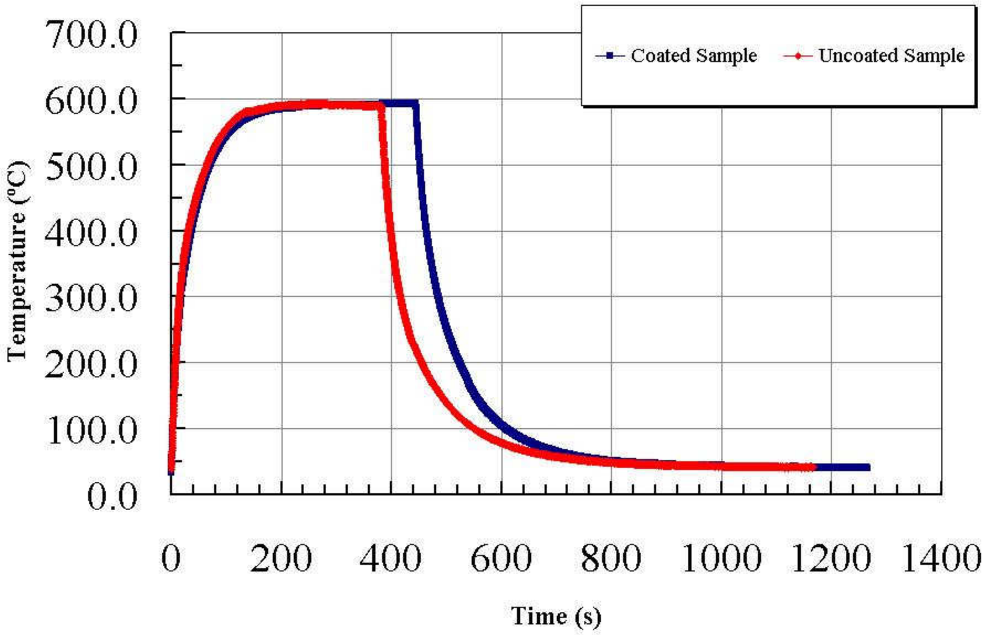

Figure 7. Heating and cooling curve for coated and uncoated sample for 0.3 combustion equivalence ratio.

Both thermal behavior for the combustion equivalence ratios shown in Figure 6 and 7, allows to see that in the same proportion as the heat transfer gradient between the sample's hot side and side in contact with atmosphere decrease, the temperature at the interface between metallic coating and the substrate tends to equalize with the temperature in contact with the hot gases from the combustion process without thermal barrier coating in a steady state heat transfer.

The difference between the thermal behavior of coated and uncoated samples can be observed checking the response time according to the heat flux input. This analyze is possible if the heating and cooling curve is parameterized conform Figure 8.

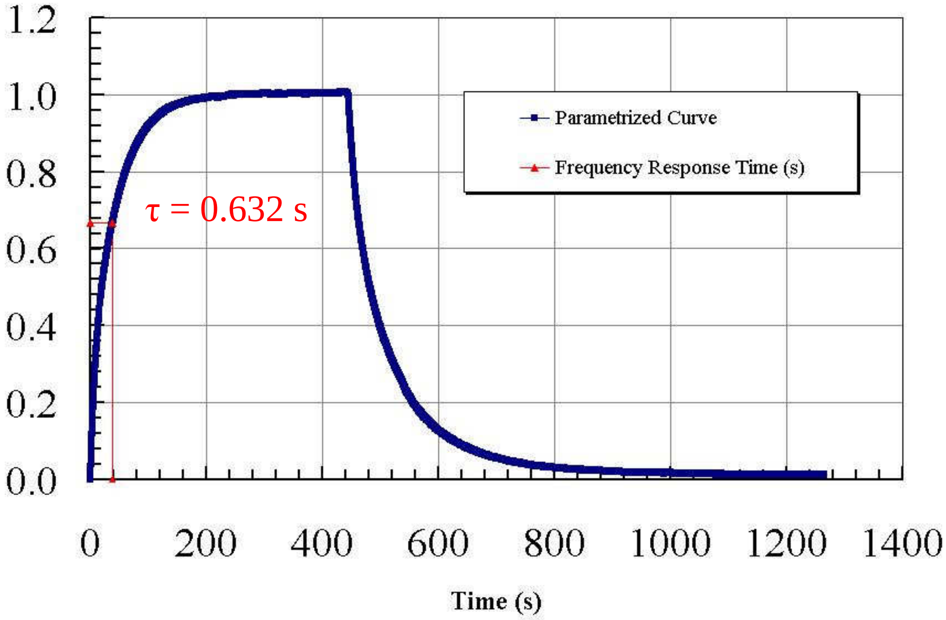

Figure 8. Parameterized heating and cooling curve for an uncoated sample.

First order systems are differential systems that involve only the first derivative in the output signal in the governing equation system. The mathematical model of these systems is shown in equation 10.

$$\tau \frac{dx(t)}{dt} + x(t) = Ky(t) \tag{10}$$

Where:
x(t) – output of the system in time t
y(t) – input of the system in time t

According Ogata, 1982, τ is time constant from a first order system. How much smaller the constant time, faster the frequency response of the system analyzed.

The response times observed for the behavior showed in Figures 6 and 7 are presented in table 2. These results were obtained observing the frequency response for a first-order system in seconds, like it's possible to see Figure 8.

Table 2: Frequency response time for coated and uncoated samples for two specific combustion equivalence ratio.

Combustion Equivalence Ratio	Uncoated Sample	Coated Sample
0.1	44.80	48.65
0.3	23.80	33.11

Analyzing the results shown from table 2 is evident that ceramics coating slows the transient state heat transfer when compared to uncoated copper alloy sample, maintaining it for longer time below the steady state heat transfer temperatures.

5. CONCLUSIONS

The thermal characterization was made with the purpose to understand the thermal behavior observing the difference in frequency response shown by coated samples, taking as starting point the fact that a thermal system be a dynamic system of first-order.

The results show a small delay in the frequency response time of the coated samples compared to uncoated samples, emphasizing that this study had the purpose to do a qualitative analyze and no quantitative analyze. One of the possible conclusions from this analysis is that the TBC maintain the surface in contact with hot gases below the steady state heat transfer temperature for a determined time.

Another important conclusion is presented in Figure 6 and 7, the necessity of a thermal gradient between the coated sample surface and the respective opposite surface of the sample, in function the trend of temperatures for coated and uncoated samples to equate when reach a steady state heat transfer.

The thermal barrier coatings replace the film cooling efficiently, which increases the specific impulse of rocket engine providing the increase of rocket's payload.

6. ACKNOWLEDGEMENTS

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