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Study about Ablative Coated Carbon Materials applied on Chambers of LRE

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The Liquid Rocket Engine - LRE are extremely complex and versatile machines, involving several areas of knowledge for your development and manufacture. The machines like Turbopumps, Reducers, Regulators, Chambers and anothers are key elements in the performance and reliability of a LRE, mainly about mass and final performance. This paper describes the preliminary research effort to determine parameters for an hypothetical ablative engine, and its consequent project, with focus on the structural design of the chamber to be built, and the selection of materials for use in the construction of this chamber. The research was started with the development of a static small combustion chamber, designed in copper alloy C11000 and stainless steel AISI 304, with a removable nozzle support, seals and nozzle inserts. The tests performed in this static chamber is intended to test the behavior of carbon material inserts, coated or not, where the parameters of wear, erosion, generation of pores and irregular wear are extrapolated for use in design of larger combustion chambers. After analyzing the test data, it was possible to identify the strength and advantages of using these materials for the manufacture of combustion chambers, and these data extrapolation was made to preliminary design of a fligh ablative thrust chamber.

Nomenclature

K	=	temperature in Kelvin degrees
TP	=	turbopump
Ve	=	gases exaust velocity
μ	=	mass relation between Total Mass of stage and Mass of stage without propellant
Isp	=	Specific Impulse
T_{00}	=	inlet temperature in turbine
DS	=	directionaly solidifield
SC	=	single crystal solidifield
ρ	=	density
L_{ad}	=	Specific Adiabatic Work
N_T	=	total power needed by turbopump
m_{gg}	=	total mass needed from gas generator to drive turbine turbopump
IPT	=	Technological Research Institute of São Paulo
APE/IAE	=	Division of Space Propulsion
AMR/IAE	=	Materials Division

I. Introduction

The design of the Brazilian Satellite Launch Vehicle - VLS01 effectively began in the 80s, based on what had been learned from the families of sounding rockets, mainly the SONDA IV. The first prototype flew in 1997, but due to a misfire in one of the engines of the first stage, it was destroyed. The second prototype flew in 1999, but by a problem during the ignition, now in the second stage, it was also destroyed. In 2003, the same occurred with VLS01v03 in Alcântara Launch Center (CLA) in Maranhão, the biggest accident of the Brazilian Space Program,

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where 21 technicians died at a premature ignition of the engines of the first stage inside the integration tower at the time in which they worked in the adjustments of pre launch of this rocket.

This was probably the moment where the need to develop and use liquid propellants was more evident throughout the program. The research here presented was developed in order to help supply this demand.

The Liquid Rocket Engines (LRE) are definitely safer systems being operated, however, such systems are also susceptible to serious accidents. The history of space programs of other nations have shown that even if operating this type of propellant, serious accidents have occurred, including victimizing tens technicians. Nevertheless, such technology has higher handling safety, beyond great increase energy efficiency. Another very important aspect is the ability to control orbit insertion quite accurate in relation to what is executed with the use of solid propellants.

The main objective of this work is to contribute to Brazil's efforts in developing technology of Liquid Rocket Engines LRE for operation in their futures Launch Vehicles.

Because the needs of new propulsion options for the vehicles of the Brazilian Space Program - PEB, several engineers and researchers it is pawning in the development of devices and components of liquid rocket engines. Aligned with these goals this study is looking for new solutions for the fast and economical manufacture of liquid rocket engine chambers, including combustion chambers and bell shape expansion section. For this important goal, the philosophies of *Responsive Space* and *Design for Manufacturing* is very appropriate in that context, the ablative technology has been showing quite viability in all the aspects and in great thrust variations.

Initially chosen, was 5kN thrust for this study, such this engine could be used for several applications, from superior launchers stages, sounding rockets, even use in large satellites or even in suborbital platforms. In the such a engine, Brazil could be apply on the last stage of the VLS-01 launcher, third stage of VLM launcher, a possible stage of the sounding rockets VS-15, VSB-30, VS-40, the Satellite of Reentered Atmospheric - SARA, and even as main engine for geostationary satellite, like SGB.

In this size of engine, in other words, 5 kN, the best combination of cost, reliability and effeteness was shown the silica fibers for the intern layer of the camera, mainly in the cylindrical section and divergence. In the point where the thermal loads and abrasion are more intense, in the throat, a material that is showing great usefulness it has been the graphite of high purity, could also be substituted with great ablation resistance increase is the tetragonal carbon/carbon, both covered with layers of ceramic films, mainly SiC and Diamond Like Carbon.

Another fundamental studied aspect is the materials used for the production of the divergence section of this camera. For applications in low altitude, the divergent can be manufactured in an integrated way in the combustion camera, directly, mainly in cases of low expansion ratio of the nozzle. Such thrust chambers manufactured only with sílica fiber they were shown low reliability and low limits of work pressure, being applied with plenty success the filament wound of those cameras with filaments of aeronautical carbon fiber, presenting like this a limit of pressure close to 7 MPa depending on the thickness of the filament wound.

II. Available Materials and Technologies

In this sense it is possible to remember some good examples of engines of that nature, beginning with the motors developed in California State University Long Beach BESNARD (2009), were they used traditional composite technology with ceramic matrix, based on carbon fiber and SiC, with capacity of temperatures in zone of 2500 K without suffering considerable erosion. In the Fig. 1 an 2.25 kN ablative chamber is shown.

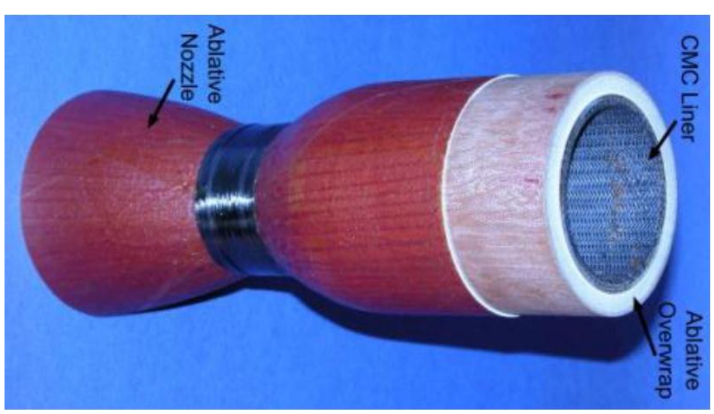

Figure 1. Ablativa chamber of 2.25 kN (BESNARD, 2009)

The development in process in the Microcosm company for your Launcher program Scorpius, is one more excellent initiative in the sense of the technologies with the philosophy Responsive Space, that in fact, had great contribution of the Microcosm experience. For so much, the company is developing two liquid rocket engines, one of 22.5 kN and another of 90 kN, both also with the technology of chamber core of silicon fiber and phenolic resin, overwrapped with carbon fiber with epoxy resin CONGER (2002), were is possible to observe in the Fig. 2.

Figure 2. Ablative engines with 22.5 kN of Thrust from Scorpius Rocket (CONGER, 2002)

It is also notable the development of the liquid engine TR108 by the company Northrop Grumman, that possesses an ablative chamber and it supplies an thrust of the order of 135 kN KIM (2005). Such chamber is manufactured in separate by the company AAE Inc, and your structure is essentially based on silicon fiber wound with phenolic resin, and quartz fiber also wound with phenolic resin in the throat zone, possessing an over lining of graphite fiber with epoxy resin as structural element, where it can be observed in the Fig. 3.

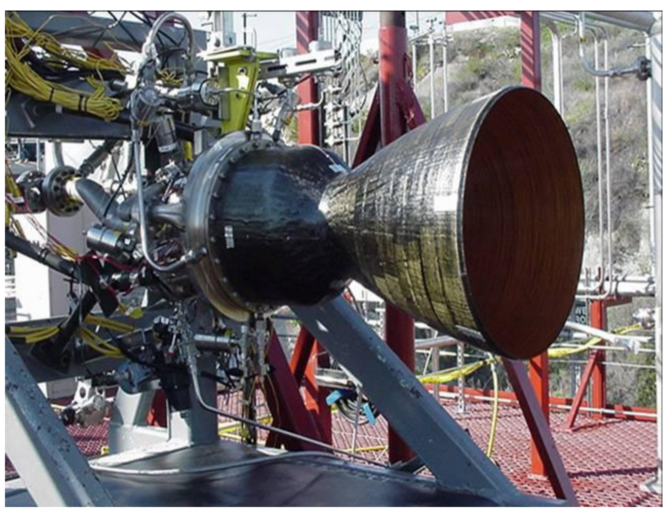

Figure 3. Ablative engine TR108 of 135kN thrust (KIM, 2005)

Several, was the initiatives to the development of this rocket engine type, it is also possible to mention the initiative of the company SpaceX when developing your first liquid rocket engine, the Merlin 1A of 378 kN of thrust, that was used in your first two launchers flights, Falcon 1, being substituted by your successor Merlin 1C of regenerative refrigeration.

Another task work in that sense was the liquid engine FASTRAC developed by Marshall Space Flight Center (FISHER, 19998), where several studies were executed, to determinate the best silicon fiber for the lamination of the interns layer of the chamber, with phenolic resin, later receiving a wound layer of carbon fiber with epoxy resin.

Such engine develops 284 kN of thrust in vacuum, and it was idealized to be a low cost liquid rocket engine, to be applied in the X-34 vehicle. In the Fig. 4 it is presented integrated and ready for your hot test.

Figure 4. FASTRAC Engine (nasa.gov)

The employed materials are essentially fibers for the internal and external layers of the chamber, resins also for both layers, and one insert of graphite coated with Silicon Carbide. In the Fig. 5, was presented the intern layer of the descent rocket engine of the American lunar module, manufactured by the company Grumman in silicon fiber with phenolic resin.

Figure 5. Internal layer in Silicon Fiber and Phenolic Resin

The external layer is manufactured in carbon fiber of aerospace employment with epoxy resin, forming this a layer of high mechanical resistance, in way to give mechanical support for the internal layer more fragile mechanically, however with larger thermal and chemical resistance. Such processing is executed in two basic ways, hand-up lamination on a concentrically mold, or wound lamination, in filament winding machine, this process being especially more efficient than the previous, because the percentage of fibers becomes larger in relation to amount of necessary resin for the lamination. This process was used in the production of the chamber of the FASTRAC engine. In the chamber proposed here, an insert will be placed in the throat, in the lamination process. Such inserted, will be EDM isotropic graphite of high purity, coated with a layer of SiC, through the process of Chemical Vapor Deposition. In this work, the cost and quality of materials determine directly the success of engine. The main materials was chosen for this application are high grade phenolic resin, high purity silicon fiber and EDM graphite, all materials with very low cost and extremely easy to machining and finishing. Next in the Fig. 6 is shown the first appearance of this materials.

Figure 6. Phenolic Resin, Silicon Fiber and EDM Graphite

III. Proposals for a New Approach

The project of gas dynamics of the thrust chamber was developed basing on Russian methodology of project of liquid rocket engines, based on KESSAEV (2008). For the injection head, was chosen the injectors of centrifuge type, bipropellant, endowing the chamber with smaller dimensions and increasing the structural efficiency of the group, being that the best choice in efficiency terms and already possessing a technological way relatively dominated in Brazil, were is not justified the employment of other injectors types. The propellants to be used in chamber are essentially variations of RP-1 and Liquid Oxygen (LOx), and the longed thrust is 5 kN. Basing on those entrance data, and in the software MathCad and AutoCad, in the Table 1 is presented the data resultants of the calculations.

Table 1. Main data of 5kN Engine

Parameter	Units	Value
Thrust	N	5000
RP-1 / LOX		
α global		2.7
Flow Mass	g/s	1428
Chamber Pressure	MPa	3
Specific Impulse (Vacuum)	s	350
Chamber Temperature	K	3600
Expansion		800
Chamber diameter	mm	105
Combustion Chamber Lenght	mm	179
Thrust Chamber Lenght	mm	627.28
Nozzle Throat	mm	33.08
Nozzle Outlet	mm	300

In a typical configuration this size of motor, the natural choice is the employment of nickel super alloys and cobalt super alloys, or even the employment of refractory alloys, like Niobium (*Columbium*). However, such choice implicates in certain penalties, the first is evidently in the total mass of the group, due the high densities of those leagues, soon after, the high cost is had due to the manufacture of those cameras, because they possess complex geometries and your materials are typically of low machinability *apud* LOUREDA (2010). To chambers of this size, the main materials are related in the Table 2 next.

Table 2. Applicable materials in small LRE

Material	ρ (g/cm3)	σ (Mpa) Yield
Inconel 600 (Nickel)	8.47	345
Inconel 625 (Nickel)	8.44	650
Inconel 718 (Nickel)	8.19	1050
C-103 (Niobium)	8.85	296
Stellite 6 (Cobalt)	8.44	700

In this version of the chamber, supposing your production in Inconel 718, the mass of employed metal should be of approximately 6,657 kg for the finished thrust chamber, demanding for your production the complete turning of a cylindrical blank of 304.8mm of diameter for 640mm of length, or the spinning of a preform. According to the Fig.7, the profile of the chamber is presented with production process by turning of a cylindrical blank, with the final geometry showed in the Fig. 8.

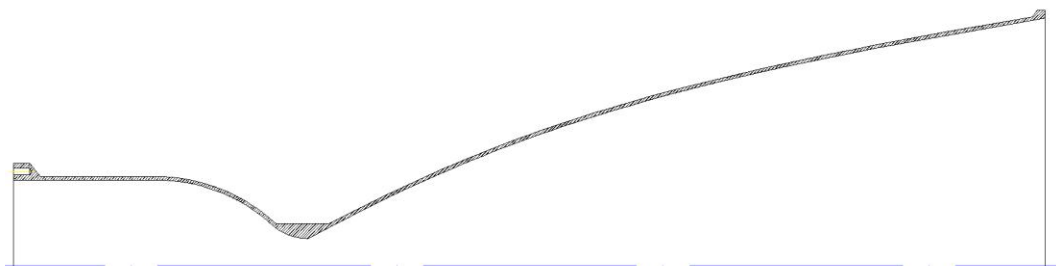

Figure 7. Metallic Geometry of a 5kN Engine

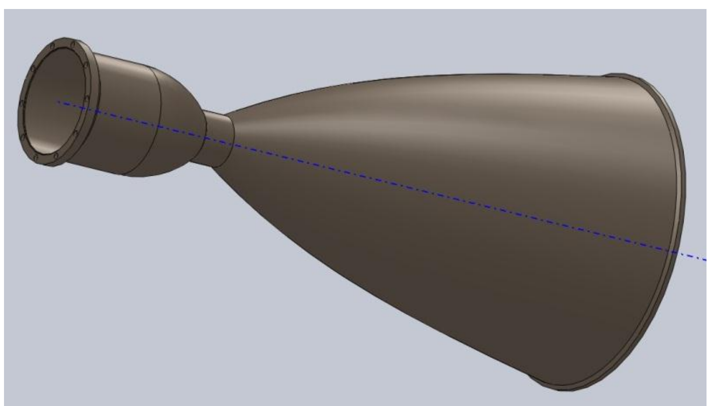

Figure 8. 3D view of the Metallic Irradiative Chamber

Then, the engines manufactured with ablative materials shown attractive options in the sense of production speed, materials cost, production cost and final mass. In this motor size, in other words, 5 kN shown the best combination in treating of cost, performance and reliability was the silicon fibers for the intern layer of the chamber, mainly in the cylindrical section and divergent, in the point where the thermal loads and abrasion are more intense, in the throat, a material that is showing a great usefulness, has been the graphite of high density, could also be substituted with great resistance increase in ablation ratio, by tetragonal carbon/carbon tetragonal both covered with layers of ceramic films presenting a considerable increment in the abrasion resistance. Another fundamental aspect studied in this work was the materials used for the production of the chamber divergent. For applications in low altitude, the divergent can be manufactured in an integrated way in combustion chamber, mainly in cases of low expansion ratio of the nozzle. Such thrust chambers, manufactured only with silicon fiber was shown low reliability and low limits of work pressure, but applied with success the wound of filaments of aeronautical carbon fiber, presenting with this pressure limits work close to 7 MPa depending on the thickness of the wound. In this thrust chambers size, with exit diameters close to the diameter of the combustion chamber, present a relationship thrust by mass quite attractive, however, in case of operation in vacuum conditions, with an expansion ratio in order of $\varepsilon = 1000$, the total mass of the thrust chamber becomes completely impracticable, because this, is necessary to choose between the penalty of excess of structural mass or reduction in Isp (v) of the engine. In alternative way to that scenery, the divergent should be quite long and thin, condition non favorable for ablative pieces, in this scenario is possible opt for the irradiative refrigeration, where the divergent should be manufactured separate from the combustion chamber, coming to be integrated later into the group. In that thrust zone, with an exit diameter three times larger than the combustion chamber, is suggested the employment of a niobium alloy for the manufacture of a divergent with bell geometry, throw the spinning machining process, arriving a thickness in the end of the divergent of 1,0 mm, being able to be integrated through a direct metallic flange in the ablative chamber. Through the employment of such materials and processes it allows to produce a thrust chamber with the mass of 0,750 kg in the combustion chamber, and 2,690 kg in the divergent section, 51% of the structural mass of a metallic thrust chamber, together with an expressive reduction of costs and time production. In the Fig. 9, the layout of proposed chamber is presented, and in Fig.10 the 3D view of the chamber.

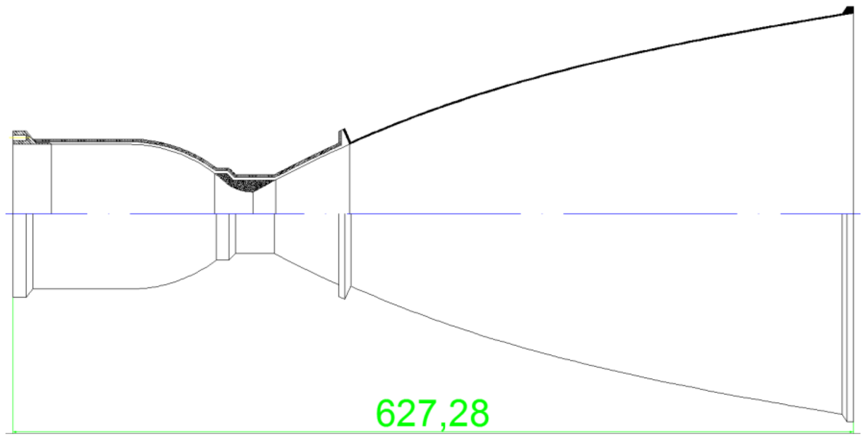

Figure 9. Ablative Thrust Chamber of 5kN

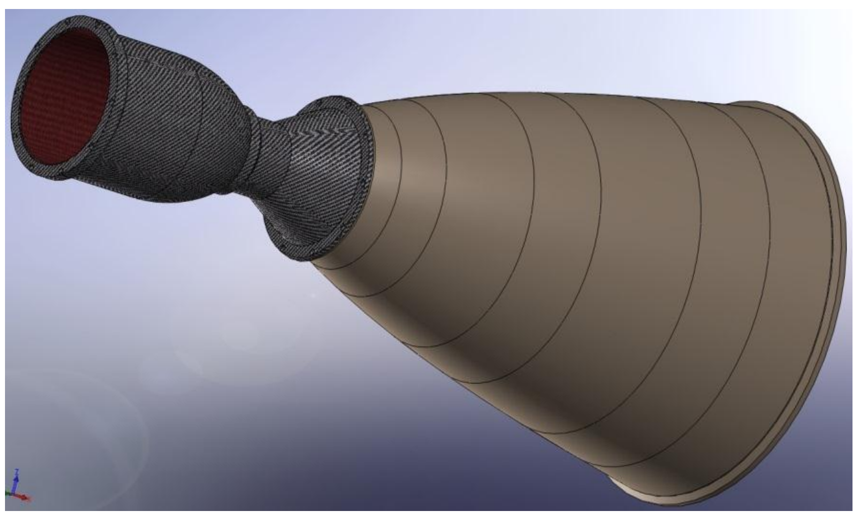

Figure 10. 3D view of the 5 kN Ablative Chamber

The manufacture of this chamber is executed through metallic molds with a division in the area of the throat, allowing the automated wound or even a mixture of wound processes with hand-up, still reducing more the production cost. The process should have the start with the positioning of the metallic flange in the extremity of the mold, and one insert of graphite covered of SiC in the region of the throat. Soon after it should be made the lamination of the sílicon fiber soaked in phenolic resin, that should be cured in temperatures of approximately 130 ^{o}C depending on the cure cycle determined initialy.

After the cure of interns layers, is followed, the lamination of the layer of carbon fiber with resin epoxy, where this new layer should have your cure cycle in greenhouse to close temperatures of 80 ^{o}C. At the end of the lamination process and it cures, the mold is dismounted and release the chamber for possible small reworks. In the Fig. 11 the detail of the combustion chamber is presented.

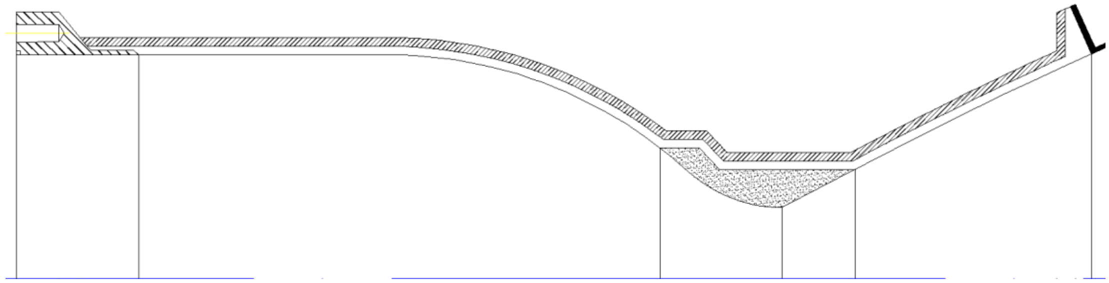

Figure 11. Ablative Combustion Chamber details

Essentially, the costs involved in the production of a thrust chamber of that size is the production of a mold with the intern geometry of chamber. This mold can be manufactured from a blank of melted aluminum, and receiving a turning for your final geometry. A mold for the chamber here researched, is estimated in approximately US$ 5.000,00. For the production of the interns layer of the chamber, phenolic resin will be used with costs near from US$ 5,00/kg, and high purity silica fiber, with cost approximately of US$ 10,00/m^2. For the external layer will be used epoxy resin that possesses approximate cost of US$ 15,00/kg and high module carbon fiber, that it possesses approximate cost of US$ 50,00/kg. With the manufactured chamber, relative tests should be accomplished, to verify the compatibility with cryogenic liquids, and after those tests begin with hot tests. During the hot test is appraised the pressure variation inside of the thrust chamber because the increase of the diameter throat of the chamber. After the hot tests, the chamber should be divided to the middle in the longitudinal direction, in way to allow the analysis of your internal layer. Through an optic microscope, the surface were suffered chemical attack will be analyzed, to determine depth of the affected area for the thermal solicitation, and the geometrical changes. Beyond of those studies, will be verified the erosion symmetry generated in the surface of the throat.

IV. Materials and Experimental Methodology

The experiments were performed in a liquid propulsion static bench, called Single Element Chamber - CEU where it was possible to simulate the environmental conditions of a liquid propellant rocket engine. The propellants chosen were ethyl alcohol and liquid oxygen (LOX), injected into the combustion chamber through a system of reservoirs and tanks, pressurized by gaseous nitrogen and gaseous oxygen respectively. The pressure in the chamber was scaled by flow mass of propellants and throat diameter of the nozzle, and adjusted in the range of 1.0 MPa The ratio used was the closest possible to the stoichiometric, with a total propellants mass flow of 25 g/s. Next in Fig. 12 shows the safety system and test bench control system , where in Fig. 13 is shown the experimental setup where the thrust chamber was fixed and tested.

Figure 12. Control system of the CEU Test Stand

Figure 13. CEU Test Stand

The main problems were the subject of this study were the interference from the ignition system, the influence of preliminary cryogenic cooling fluid and combustion inside the chamber over the life of the carbon throat insert. In order to assess this exposure, the static test chamber was designed with a removable throat, where it could be replaced by another pieces of different materials and geometries. In this study, the inserts were assessed was manufactured by a extra fine grain graphite, uncoated and Silicon Carbide coated with two different thicknesses. In fig. 14 below the tested inserts are presented.

Figure 14. Carbon Nozzle Inserts

The chamber designed to test these inserts was produced in copper C11000 for its thermal conductivity and the rest of components, requiring greater mechanical strength were manufactured in AISI 304 stainless steel, including injection head, ignition system and nozzle holder. Below in Fig. 15 the projected chamber is presented.

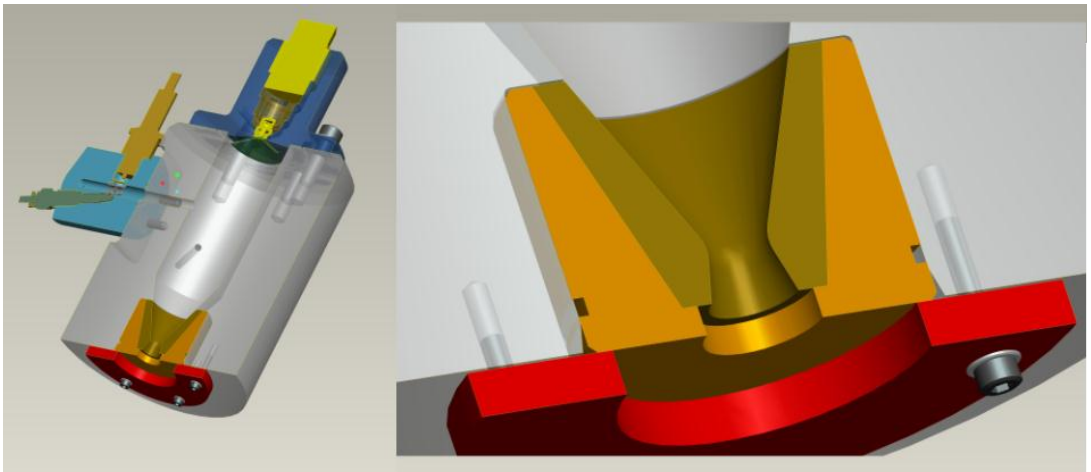

Figure 15. Static Test Chamber

V. Conclusions

Through the employment of chamber as described in this work, is possible the drastic reduction of cost of production of liquid rocket engines, as well as increasing the production speed sensibly. With the construction of liquid engines with ablative chambers, it is also possible the considerable reduction of the mass of the engine. In that case, an engine of that size suffered a reduction of mass of the order of 100% of the structural mass of the chamber. Supposing your application in last stages of launch vehicles, that mass turns directly into addition of useful payload in orbit.

Based on these studies it was possible to design the last thrust chamber based on composites and thin divergent from inconel or niobium for vacuum applications, or without such divergent in atmospheric flight conditions, as shown in Fig. 10. The use of such materials in the manufacture of liquid rocket engines have been shown to be feasible in recent decades and has been reinvented with new philosophies like Responsive Space, Design for Manufacturing and the use of COTS.

The Static test chamber has proved quite robust and very practical for testing with different flows, pressures, methods of ignition and mostly for different geometries and materials for nozzles.

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