

Ablative Heating Technology in Hypersonic Re-entry Vehicles and Cruise Aircrafts



Ali Asgar, Sarath Raj N.S, Jerrin Thadathil Varghese

Abstract— The aim of researches conducted in thermal protection systems in aeronautics and astronautics field of engineering is to generally defend the craft from high heat loads during operation while operating at hypersonic regimes in air and space. The motive of the following composition is to draft a review analysis on ablative heating materials as thermal protective equipment on reusable planetary/atmospheric re-entry vehicles such as a space shuttle, an inter-continental ballistic missile, or a hypersonic cruise missile. The heat liberation can cause much damage to the aircraft/spacecraft whilst operation which is generally beyond repair. It is therefore of utmost importance to research multiple strategy to reduce the effect of shockwaves damage to spacecraft/aircraft materials. We shall initiate the analysis by mentioning some re-usable tile thermal protection system types such as high temperature reusable surface insulation tiles (H.R.S.I), fibrous refractory composite insulation tiles (F.R.C.I), low temperature reusable surface insulation tiles (L.R.S.I) and gradually move on to ablative thermal protection systems with the advent of reinforced carbon-carbon's application in astronautics and aeronautics respectively.

Keywords— Adiabatic compression, Radiative heat transfer, Emissivity, Planetary re-entry, Hypersonic compressible flow regime, Shock waves, Boundary layers, Pyrolysis, Ablative heat shield technology

I. INTRODUCTION

Planetary re-entry is always pictured as meteors motioning towards Earth with a bright light tailing behind them. These meteors are actually meteoroids that somehow entered the Earth's atmosphere and started drowning in [1]. It is commonly believed however that the cause of this immense heat liberation is due to friction caused by the planet's dense atmosphere and pressure [2]. Friction however, does change some of the aggregate thermal entropy [3]. Following what we see is what we call in popular culture as a 'breaking star'[4]. Through advancements in meteor physics and shockwave physics we now know that the cause of the liberation of heat and light in meteor fall is not because of friction due to planet's dense atmosphere and pressure, but because of adiabatic compression of air in the atmosphere and the formation of shockwaves [5].

All shockwaves are associated with supersonic and hypersonic vehicles [6]. This includes all spacecraft that are entering a planet with a dense atmosphere and all aircrafts moving with hypersonic speeds. It was world war II when the pilots engaged in long vertical speed dives noticed several irregularities in flight [7].

Their lifts decreased and aerodynamic drag increased significantly. It was at this time the pilots realized that supersonic flight was impossible [7]. It was wrong however. It always seemed impossible for the early space exploration societies to ever send a vehicle out the atmosphere of Earth and enjoy interstellar photographs. It was also due to the fact that the re-entry of the spacecraft back into the Earth's atmosphere again was impossible due to potential hull loss of the space vehicle due to immense heat loads while re-entering the planet [8]. Now we know that shockwaves are responsible for it [9]. The control of shock waves were necessary and the amount of literature is evident on the subject of matter. Thus, in 1941, Regenscheit studied the effects of shockwaves on a supersonic (or transonic) aircraft by cutting out a slit on the aircraft's airfoil[10]. The same apparatus was used by Fage and Sargent in 1943 to similar conclusions. This evolved into a refined area of study afterwards[11].

A. Generalized introduction to hypersonic flows

$$M_{\alpha} == \frac{U_{\alpha}}{a_{\alpha}} \gg 1$$

In the above equation, M_{α} represents the Mach number while U_{α} and a_{α} represent velocity of the vehicle and speed of sound respectively (where the limiting case in which the Mach number reaches infinity because the free stream velocity reaches infinity whereas the mean thermodynamic state of the free stream remains constant). The type of speed the vehicle possesses clearly depends on the Mach number of the vehicle. Transonic vehicle is said to be a transonic aircraft/spacecraft when the Mach number is exactly 1. whereas for supersonic vehicle, the Mach number is usually more than 1. similarly, subsonic aircrafts are less than 1 and more than 0. Now, hypersonic aircrafts or other re-entry space vehicles/ space debris (meteoroids) possess velocities more than 4. In such conditions of high velocities, the spacecraft or the aircraft is subjected to soaring heat loads. The false fact that friction causes the heat elevation is a misnomer. There are two types of heat transfer mechanisms involved in the heating up of the spacecraft during operating in hypersonic speed regimes. They are:

- Convective Heating. It again has two main sources: i) ionized hot gas passing through the body and ii) catalytic combination reactions between the surface of the spacecraft and the surrounding atmosphere [12].

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- Radiative Heat Load. The only dominant source of radiative thermal load is the associated shockwave with the craft operating at hypersonic speed limits. (Like an ordinary wave, a shock wave carries energy and can propagate through a medium but is characterized by an abrupt, nearly discontinuous, change in pressure, temperature and density of the medium, causing the asymmetric disturbance to propagate through the atmosphere producing sonic booms[13][14]).

As velocity is increased, both conservative and radiative heating starts increasing. At very high velocities, it is seen that radiative heat shall always come to quickly exercise control over the convective heat fluxes, that's because the non-linear relationship between convective heat fluxes and radiative heat fluxes with respect to velocity shows that the ratio is much stronger in respect to velocity in case of radiative heating. Radiative heating, thus, is seen predominating in the posterior phases[12]. For earth, atmospheric entry occurs at the Kármán Line at an altitude of 100 km (62.14 nautical miles) above the mean sea level. Free-falling objects reach extreme velocities while gaining momentum through space in-line towards Earth due to the effect of its own gravity, and are restricted in motion by friction (though, a misconception, still prevalent amongst majority of the population) upon crossing the thresholds of Earth's atmosphere. Meteors are also often seen penetrating quite as speedy relative to the Earth simply because their eccentric orbital paths are different from that of the Earth before they are influenced by the Earth's magnetic and gravitational field. Most controlled objects enter at hypersonic speeds because of their orbits, at a point, intersects with the atmosphere of Earth or in-fact, initiate their flight on the surface of Earth (Inter-continental Ballistic Missiles re-entry vehicles), orbital (The Soyuz space station), or extra-terrestrial bodies of unbounded trajectories (small meteoroids). Research is still carried out at all levels to enable travel to and from atmospheres of high densities, that is, in case of space re-entry shuttles and intra-atmosphere cruise aircrafts. An alternative low velocity method of controlled atmospheric entry was also taken into examination and it is called 'buoyancy'[15]. Early re-entry vehicle concepts were visualized in advanced optical method analysis of high speed wind tunnel tests. However, the abstraction of ablative heating was described as early as 1920 by Robert Goddard: "In the case of meteors, which enter the Earth with speeds as high as 30 miles per second (48 km per second), and the interiors of the meteors remain cold, and the erosion is due to a large extent, chipping or cracking of the suddenly heated surface. For this reason, if the outer surface of the apparatus were to consist of layers of a very infusible hard substance with layers of a poor heat conductor between the surface would not be eroded to any considerable extent, especially as the velocity of the apparatus would not be nearly so great as that of the average meteor" [16]. Then gradually this technology was pioneered by H. Julian Allen and A. J. Eggers Jr. of the National Advisory Committee for Aeronautics (N. A. C. A). at Ames Research Centre [17].

II. INSULATIVE AND REUSABLE THERMAL PROTECTION TILES.

A reusable thermal protection system should not only be lightweight, but durable, operable and cost effective. These are few of the technological parameters required by most of the reusable launch vehicles such as a space shuttle upon re-entry.

The first wave of thermal protection systems were chosen to be reusable insulators such as those on current space shuttles. The first insulative, reusable type of thermal protection system were in the form of ceramic tiles. Ceramic tiles such as alumina enhanced thermal barrier (A.E.T.B) with toughened unpiece fibrous insulation (T.U.F.I) and reaction cured glass (R.C.G) coatings, have been developed to be significantly stronger and more resistant to precipitation erosion than the previous premium ceramic tiles.

Tailorable advanced blanket insulation (T.A.B.I), also developed by N.A.S.A Ames is being proposed as a cheaper, more easily integrated and installed replacement for tiles over large areas of future vehicles [18].

1) High temperature reusable surface insulation tiles (H.R.S.I).

These tiles are made from low density, high purity silica and 99.8% amorphous fiber (derived from common sand). Since 90% of the whole tile is air therefore, the remaining 10% is material. H.R.S.I weighs around 9 pounds per square foot. The manufacturing process includes preparing a semi-liquid heterogeneous mixture containing fibers mixed with water which is then casted to form soft, porous bricks to which a colloidal silica binder solution is added. After being compressed and cut to form the desired shape. The thickness of these tiles vary from spacecraft to spacecraft. Normally the thickness of these tiles vary from an inch to five inches. Again, the thickness is varied from vehicle to vehicle depending upon the operation complexity and heat encountered upon reentry. There are specifications as to how these tiles be manufactured and quality control is a necessity. The tiles go through a lot of atmospheric conditions in extremities like extreme cold to scorching hot as space does not have atmosphere. Tiles are pre-checked to make sure they do not break or crack. Also, H.R.S.I tiles are supposed to made waterproof by adding a coat on top and sides with a glassy material and a liquid carrier. These are then baked at an ambient temperature of 2,300 degree Fahrenheit. As the tiles undergo extreme temperature variations, these tiles expand and contract. Therefore, a gap of 0.0025 millimeter to 0.065 millimeter is left then these gaps are filled with a material called as a Nomex felt, known as filler bars [19][20].

2) Fibrous refractory composite insulation tiles (F.R.C.I) tiles.

These tiles are just a new form of H.R.S.I tiles which are high strength tiles derived by AB312 Columbia Borosilicate fiber called Nextel, which is added to the pure silica tile semi-liquid heterogeneous mixture. As the tile's silica slurry is being sintered (compressed), the Nextel activates boron fusion and, kind of, welds the pure silica fibers into a rigid, solid structure. As this tile is 20% Nextel and 80% silica, the tile has different physical, chemical and thermodynamic properties than its predecessor tile, H.R.S.I. It's evident as these tiles possess 3 times more tensile strength than the original H.R.S.I tiles, and also, these tile can be, moreover, be used at 300 degree Fahrenheit more than that of its predecessor, the H.R.S.I tiles[21][20].

3) Low temperature reusable surface insulation tiles (L.R.S.I).

L.R.S.I have the same construction as that of the premium H.R.S.I silica tiles and also serve as a basic function as that of the 99.8% pure silica H.R.S.I tiles except that they are 8x8 inches and are optically and water resistant (0.10 inches thick). The outer coating is made of silica compounds with satiny aluminum oxide to obtain the optimum optical properties ever[22][20].

4) *Advanced flexible reusable surface insulation blanket (A.F.R.S.I).*

0.01 millimeter to 0.002 millimeter thick coating, is a low density fibrous silica batting that is composed of a high purity silica and 99.8% amorphous silica fibers. The batting is compressed between a woven high temperature silica fabric and a low temperature glass fabric. This compressed batting is now sewn together with silica thread, it now has a quilt like appearance. Density, though, generally depends upon the types of mission and ambient varying temperature upon operation and planetary reentry's ambient temperatures, the average density remains between 8 pounds per cubic feet and 9 pounds per cubic inches and also varies in thickness between 0.45 inches to 0.95 inches[23][20].

5) *Fell reusable surface insulation (F.R.S.I).*

Varies in thickness from 0.16 inch to 0.40 inch, again depending upon the type of mission and total heat load encountered upon reentry and type of operation. This kind of insulation is installed on to the aircraft by silicon derivative adhesive at about 0.20 inch. There should be waterproofing, optical and thermal conducting properties by coating the tiles on the top and sides by a white silicon elastomer to achieve the require resistant properties. This one of the most dominant insulation used in the reentry spacecraft as it covers nearly 50% of the space shuttle's body[24][20].

6) *Reinforced Carbon-Carbon.*

The carbon-carbon composites are a well-defined species of new emerging engineering materials that are ceramic in basic essence and complexion, but they exhibit properties wide ranging from brittle to pseudo-plastic behavior. The carbon-carbon is also known as inverse composite due to carbon fiber embedded in the whole carbon matrix, therefore we can draw a close conclusion that carbon-carbon is an idiosyncratic all carbon composite material. And due to their outstanding thermos-structural properties, the reinforced carbon-carbon are employed in specialized field of applications such as reusable spacecrafts (space shuttles) or any kind of reentry or hypersonic cruise operations such as the intercontinental ballistic missiles or the hypersonic medium to large range cruise missiles. In reentry vehicles they are applied in the nose tips, leading edges and rocket nozzles. And aircrafts, they are used in braking discs. Such is Airbus A320. The multi-directional carbon-carbon technology in versatile and does offer a great deal of design flexibility.

Development of carbon-carbon products involve a process using the liquid (phenol resin) impregnation process. The prime reasons why it is chosen to be a reliable engineering material is because of a few various reasons, primarily because it doesn't melt even at temperatures above 3000 degree Celsius, it maintains an exceptional frictional properties [25].

The reinforced carbon-carbon is produced by impregnating carbon containing material such as rayon cloth with a phenolic resin. The cloth is filled with some semiliquid material. After being placed in an autoclave for curing, the cloth is then pyrolyzed to convert the containing

resin to pure carbon. After its treatment in alcohol, it is pyrolyzed again to form pure carbon. The process is then repeated till the purest form of carbon-carbon material is obtained. The material is then coated with silicon carbide to prevent it to oxidize. Then again the formed material is kept in furnace/ baked with alumina at an ambient temperature of 3200 degree Fahrenheit to reinforce silicon-carbon bonds within the structure to emerge and further protect the to corrode due to oxidization[20].

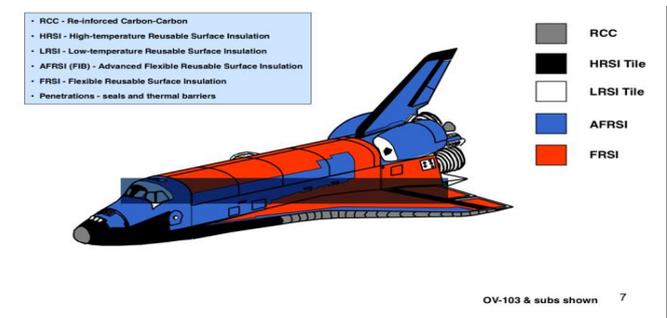


Fig. 1.Orbiter TPS Configuration [20]

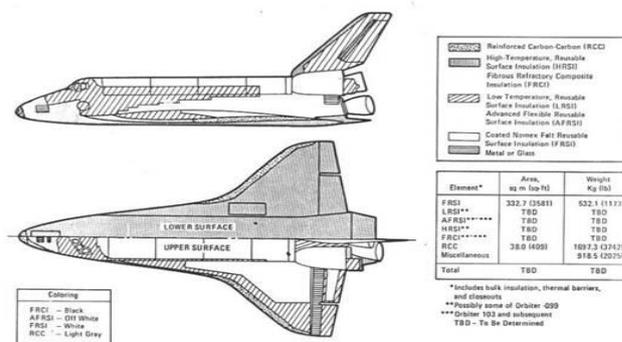


Fig. 2. Figure Name [26]

III. TOUGHENED UNI-PIECE FIBROUS REINFORCED OXIDATION RESISTANT COMPOSITE (T.U.F.R.O.C)

This composite was designed by D.B. Leiser and D.M. Stewart Jr., in 2007 in NASA's Ames Research Centre in Moffett Field, California and this patented invention was an award from NASA in Ames Research Centre to be the best thermal protection system for the leading edge area of the re-entry shuttle during the atmospheric re-entry phase. This was primarily tested on the X-37B space shuttle in the same year in October 2007.

T.U.F.R.O.C is an extension in thermal protection equipment from the reinforced carbon-carbon shield. It originally was designed for the leading edge areas of the re-entry craft and other control surfaces can also be protected using this type of fibrous insulation technology.

The composite consists of a toughened, high temperature surface cap and a low thermal-conducting base and can be used on both sharp and blunt leading edge control surfaces of the vehicle [27].

This system of fibrous insulation-ware comprises of R.O.C.C.I (refractory oxidation resistant ceramic carbon-insulation), which equips the craft with dimensional stability to the outer mold line, while the fibrous base material of the T.U.F.R.O.C resin provides maximum thermal insulation for the craft structure.



The composite has graded surface treatments applied by impregnation to both top and bottom including the cap and the base.

Using the T.U.F.R.O.C composite instead of the normal reinforced carbon-carbon is more advantageous as the reinforced carbon-carbon is 100 times more expensive than T.U.F.R.O.C. The normal leading edge thermal protection equipment weighs around 1.6 grams/cubic centimeter, but the T.U.F.R.O.C weighs only 0.4 grams/ cubic centimeters [28].

A. Thermal Emissivity.

We know that the re-entry spacecraft and hypersonic cruise aircrafts (cruise missiles) travel at hypersonic speed regimes allowing it to form a boundary layer around the body touching directly to its skin and including a saturation point depending upon the direction of travel and density of atmospheric elements. And on top of the boundary layer, we have the shock layer formed due to convective and conducting heat transfer cycle prevalent due to its virtue. Due to these heat transfer cycles, the body starts radiating energy in the form of heat, this heat is liberated in the form of electro-magnetic radiations. "Emissivity" of the surface of the material is known as its efficiency in emitting energy in the form of electro-magnetic radiations. Generally in re-entry spacecraft, this radiation is in the form of thermal radiation. This may include the parts of electro-magnetic spectrum visible to the naked eye (the trail of light we see when we look at a re-entry capsule during its descent down the atmosphere, example, meteors and comets). And also in the form of infrared radiations that we cannot see with our naked eyes. Quantitatively speaking, "Emissivity" is the ratio of thermal radiation from the surface of any ordinary body to the radiation from any perfect black body material (theoretical) material at its thermodynamic temperature 'T'.

Black body radiant emittance is given by:

$$j = \sigma T^4$$

Where,

$$\sigma = \frac{2\pi^5 K^4}{15C^2 h^3}$$

where, K is the Boltzmann's constant, C is the speed of light in vacuum and h is the Planck's constant.

Where calculated the value of $\sigma = 5.67 \times 10^{-8} W/m^2 K^{-4}$

The knowledge of thermal radiant emissivity is used in thermal shielding of reusable hypersonic re-entry spacecraft and intra-atmospheric cruise aircrafts. There high emissivity coatings with emissivity values near 0.9 are coated on the surface of insulating ceramics [29]. These facilities cooling of the spacecraft and protection of underlying structural payloads in single entry capsules.

Hemispherical emissivity is given by:

$$\epsilon = \frac{M_e}{M_e^o}$$

M_e = radiant exitance of surface of the TPS

M_e^o = radiant exitance of an ideal black body

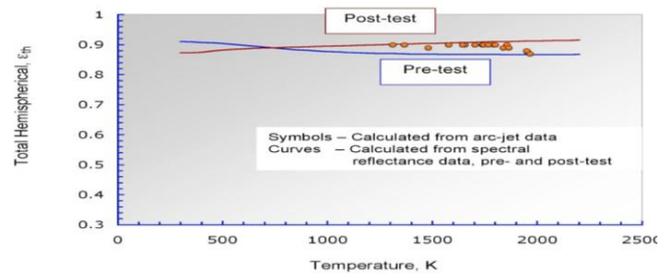


Fig. 3. Figure represents the variation of thermal emittance ϵ_{th} of TUFROC substance with increasing temperature [27]

IV. ULTRA HIGH TEMPERATURE CERAMICS (U.H.T.C.)

These are one of the most important engineering material groups known to us. The ultra-high temperature ceramics were first discovered in around late 1960s and early 1970s. these materials are generally used in the aerospace department widely due to their ability to stay in-tact even in the most hostile situations during interstellar operations. The U.H.T.Cs operate on ambient temperatures of more than 3000 degree Celsius. These classification of specialized engineering materials include carbides, nitrides and most importantly, borides. The first ultra-high temperature ceramic manufactured was silicon carbide. Nowadays, most commonly used ceramics of these kinds are zirconium diboride and hafnium diboride, both consisting of the boride family. The early amount of work done in this field is given by Fenter [30]. Even before the need to deploy materials in the hypersonic aerospace industry, these materials were widely used in other manufacturing products such as ball bearings, armors for armored vehicles and even turbine blades for supersonic jets. There are more than 300 materials with their melting points well beyond 300 degree Celsius. But it's always good to have more choices as in engineering sciences and practices it's not the one factor we are looking for. We might need a combination suited well for our purpose, design and type of operation. For example, most engines and all leading edge applications will involve exposure to harmful refractory oxides and shall undergo oxidation to form some or the other form of corrosion material to degrade performance of the spacecraft or the aircraft in operation. However, in this paper, we are only interested in the subject's thermal shock capacity. The results of thermal shock is summarized below table and figure [31]

TABLE I. THERMAL SHOCK OF UHTCS

Specimen Number	Material	O/F	Target Temperature, °C	Exposure Time, s	Observations
N1382	Si3N4(AS-800)	1.7	1327	5	survived, minimal silica flow along surface
F6923	ZS	1.7	1327	5	cracked during cool down
F6924	ZS	1.7	1327	5	survived*, minimal oxidation mostly at hot spot
F6925	ZS	1.7	1327	210	cracked*, significant oxidation on surface
F6926	ZS	2.3	1627	180	significant material loss, half of sample broke away*
F6913	ZSC	1.7	1327	5	survived*, minimal oxidation on surface
F6914	ZSC	1.7	1327	75	survived*, significant oxidation on surface
F6915	ZSC	2.3	1627	180	cracked, significant material loss due to spallation*

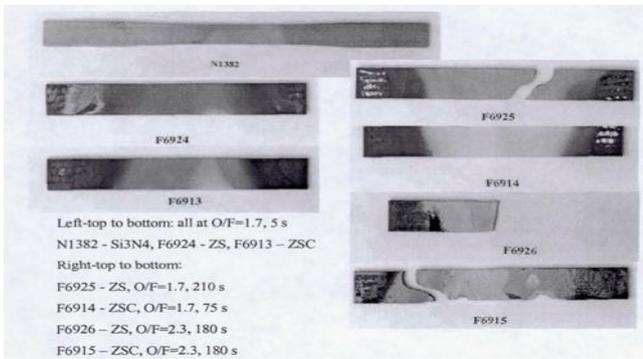


Fig. 4. Photographs of specimen tested in thermal shock

V. ABLATIVE THERMAL PROTECTION SYSTEM.

A. Introduction

Ablative thermal protection systems is the most researched thermal protective system in the aerospace industries worldwide due to their flexibility of being and simplicity of performance. The basic idea behind the ablative heat shield is that the material adopted is directly exposed to the boundary layer just above the surface of the skin. The material is allowed to withstand temperatures beyond a limit where all or most of the materials known to man starts melting into a hot lava sort of form (this ablation process occurs majorly at two levels, first, the material melts and then sublimates into gas and secondly, the inner bulk of material undergoes pyrolysis which is the central idea behind the ablative heat shielding technology). These materials, however have decent thermal load bearing capacities. When they encounter soaring heat loads above their limit of tolerance, they start pyrolyzing, form into a hot glowing liquid and evaporate, into the environment around the spacecraft where entropy is already at its maximum. The amount of material mounted at the base of the capsule is calculated well in advance before the initiation of the operation and installed on the base of the capsule.

B. Phenolic Impregnated Carbon Ablator (P.I.C.A).

The phenolic impregnated carbon impregnator is a part of a group of materials known as lightweight ceramics ablators which were originally used in former Apollo and Viking missions for outer space discovery missions program to investigate lunar soils and mars from its orbit and to land consequently. Phenolic impregnated carbon ablator was first discovered in 1980's and was first used in Stardust robotic space probe spacecraft, which was first launched on 7th February, 1999. The aim of the stardust was to collect dust samples from the far comet in coma, the wild 2. The whole mission lasted for a total of 12 years, 1 month and 17 days. On 15th of January, 2006, the sample return capsule detached from the main ship (Stardust space probe) at around 05:57 UTC. Then the sample collection and return capsule initiated re-entering process at around 09:50 UTC [32]. The sample return capsule was then observed to reach a maximum velocity of up to 12 kilometers in one second. This was the highest observed speed encountered by mankind in outer space missions and in general [33]. And furthermore, the robotic probe reached from 36 Mach number in compressible ambient flow regime to subsonic flow regime in just 110 seconds [34].

It's a wonder how the craft survived the searing heat loads at speed regimes well above hypersonic compressible

flow regime. That was the time when the phenolic impregnated carbon ablator was given its desirable credit. All lightweight ceramic ablators use very thin carbon fibers mixed or hybrid well with organic resin that is allowed to infiltrate the carbon material. The pre-requisite carbon fibers are supplied from a fixed (monopolized) manufacturer (Fibre Materials, Inc.). The carbon fibers are commercially known as Fiber form Insulation. The latter known fine materials are chopped specially up to 14 micrometers to 16 micrometers (14×10^{-6} meters to 16×10^{-6} meters) and total length up to 1600 micrometers. A water soluble organic resin is added in to the water with the fiber form insulation carbon fibers and slurry is formed. Heat treatment of soaring high temperatures is then performed on the slurry. The temperature raise can be soared as high as 1440 degree Fahrenheit to 3240 degree Fahrenheit. The water soluble organic resin used in the slurry is a phenol-formaldehyde resin formed through polycondensation reactions between phenol and formaldehyde hydrated solutions in the presence of necessary and suitable foreign elements to act as catalysts. The whole reaction can be completed in 1 or 2 stages.

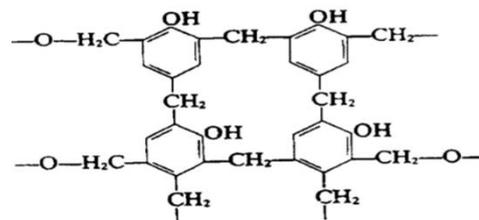


Fig. 5. Molecular Structure if cured phenolic resins

Necessary tests are performed at the initial to verify the subject material's productive capabilities and its liability in the performed missions. Failure to check and then approve the test subject material may lead to disastrous or worst catastrophic failures leading to economic (financial), government, civilian, expertized personnel life's losses and overall mission fiasco.

C. Thermal conductivity tests.

This would provide an obvious experiment result as the fiber form carbon fibers were initially manufactured to aid insulative facilities for vacuum environments and also high temperature scalping and roasting purpose industrial furnaces. Though, mathematically it can be deducible that the thermal conductivity of the material can be increased with an increase with the increasing temperature as the same that can be expressed in the following Arrhenius function:

$$K = Ae^{BT}$$

Where:

K is the thermal conductivity.

A is a material dependent coefficient (or thermal conductivity at 32 degree Fahrenheit).

B is also a material dependent constant how fast the thermal conductivity increases with ambient rising temperature.

T is the mean ambient aerial temperature.

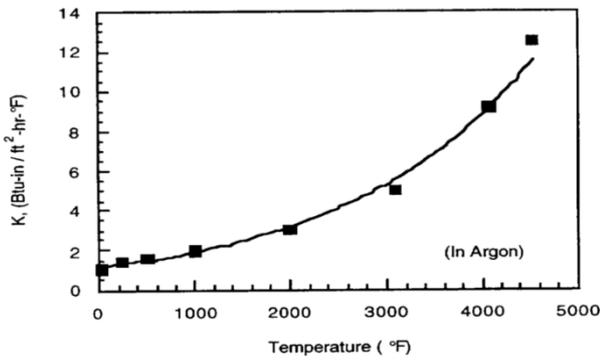


Fig. 6. Thermal conductivity of Carbon Fiberform

D. High enthalpy compressive flow regime environment.

In the United States, the hypersonic aircraft and launch vehicle tests are performed at N.A.S.A Ames research center based in California. The arc-jet testing facility located at N.A.S.A's Ames research center is exclusively built to carry out artificial hypersonic environments to test thermal shield material products. Such an arc-jet testing facility at Ames includes 60 megawatt interaction section. It uses a very high voltage to initially heat up the gas in the 3.15 inch constrictor column. After the electric discharge, the heated gas expands at supersonic speeds through a convergent-divergent conical nozzle whose outlet diameter is 13 inches at exit. The air then is passed in the evacuation chamber where the test specimen is kept on a swing arm. The resulting reaction between the flowing winds at above sonic speeds and the stationary test specimen create environments similar to space to atmosphere re-entry conditions. The interaction heating facility can simulate exit enthalpies nearing 19,500 Btu/Ibm and tangential velocities up to Mach 7. The final outcome of these performed experiment always depend on initial variables. The arc-jet experiments help figure out material's main reaction towards severe heat loads and maximum entropy environments encountered by the re-entry launch vehicle or any hypersonic cruise missile or any inter-continental ballistic missile up on reaching unstable and highly volatile hypersonic compressible flow regimes. Variable like heat fluxes, calibrations tests and stagnation pressure tests are performed. Mathematical analysis show that the total heat flux passing throughout the stagnation point initiating at the nose of the aircraft show that dependent on the craft's nose radius. This is demonstrated through the simplified Fay-Riddell equation:

$$H_{hemi} = C \sqrt{\frac{P_{t2}}{R_n}}$$

Where,

H_{hemi} is heat flux

C is aggregate eat transfer coefficient

P_{t2} is the stagnation pressure

R_n is the nose radius

The effective heat of ablation of the material is as follows:

$$H_{eff} = \frac{q_{cw}}{s\rho_1}$$

Where,

H_{eff} is the effective heat of ablation

q_{cw} is the cold point wall stagnation heat flux.

s is the recession rate.

ρ_1 is the density of the virgin PICA sample.

E. Standard PICA test models.

Test results are divided into three different regime categories based on encountered and effectively measured heat fluxes due to varied radiation mode of heat transfer.

- 1) The first regime ranging from heat flux encountered 300 to 400 Btu/ft square second.
- 2) The second regime ranging from 400 Btu/ft square second up to 1400 Btu/ft square second.
- 3) The third regime of varied heat fluxes encountered ranging from sublimation rate controlled rate of above 1800 Btu/ft square second.

F. Thermogravimetric analysis.

The thermo-gravimetric analysis are performed over an object to observe the physical and chemical change of the test subject under control over a gradual or drastic change of temperature over a certain time differential. This serves as an aid to detect and analyze thermal response and ablation characteristics of PICA (mainly PICA-M2-7) which are tested at 1065 Btu/ft square seconds. The method of examination included to calculate the percentage of phenolic resin content remaining from the control sample under experimental examination. Phenolic resin amount in percentage at each different sections of the sample were calculated based on the mass lost from the original sample and the criteria to evaluate the results were based on the given mathematical model:

$$Re(\%) = \frac{M_r}{M_p} \times 100$$

Where,

M_r is the lost mass of the sample as whole.

M_p is the mass of pyrolyzed phenolic resin.

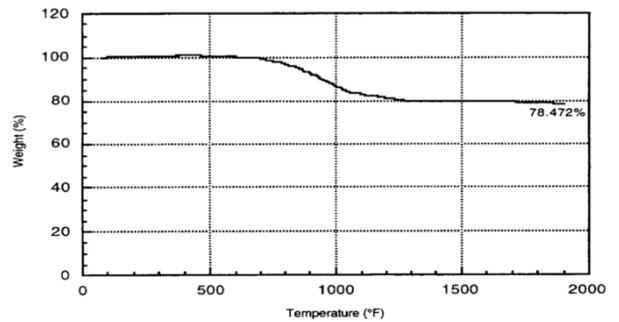


Fig. 7. TGA of PICA in Argon

G. Infrared (I.R), ultraviolet (U.V) spectroscopy.

The samples were also tested in high radiation environments as it is obvious during high entropy environment of adiabatic compression taking place during atmospheric re-entry phase of a launch vehicle, the radiative heat transfer rate dominate over convective heat transfer cycle. Therefore, it is of most importance to test the control sample under examination to test it under high radio noise environment.

A Perkin-Elmer Lambda-9 spectrophotometer was used during the experiment to measure bandwidth wavelengths ranging from around 0.25×10^{-6} m to 18×10^{-6} m. Measured readings were then converted to emittances figures for the ease of conversions. The following mathematical treatment was used to perform the same:

$$\varepsilon(\lambda, T) = 1 - \rho(\lambda, T_R)$$

The above expressions has its own shortcomings and can not be worked with efficiently. This expression is only computable with situations or pure ambient environments where the sample object under control is possessing diffusely radiating opaque surface with thermal equilibrium with its surroundings in a constant temperature (T_R).

Hence forth, the temperature dependent function of emittance is in its integral form over Planck's distribution function at temperature T.

$$\varepsilon(T) = \frac{\int_{\lambda_i}^{\lambda_u} \varepsilon(\lambda, T_R) e_{\lambda b}(\lambda, T) d\lambda}{\int_{\lambda_i}^{\lambda_u} e_{\lambda b}(\lambda, T) d\lambda}$$

This is Planck's distribution and was used to calculate emissivity in the sample object under control, which shown in the figure below.

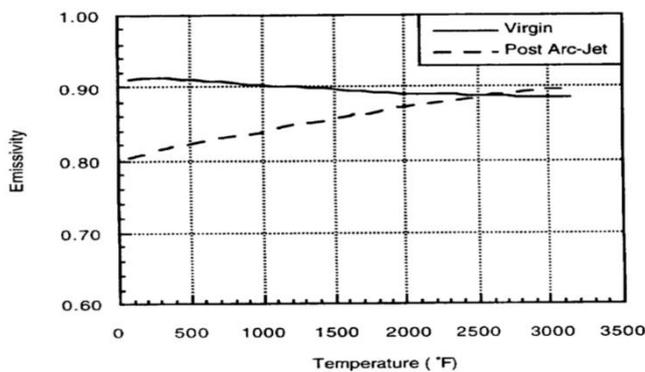


Fig. 8. Emissivity of virgin and charred standard PICA.

VI. CONCLUSION

It is well evident that significant research has been conducted since 1940's onward on the field of thermal protection systems to protect the launch vehicle from soaring heat loads due to its operation at hypersonic flow regimes. These thermal protection equipment can be used on any aerial or space operating vehicles approaching high velocities capable of forming high pressure boundary layers and shock layers depending in flight maneuvers and flight paths. From the figures provided from the arc-jet experiments on various ceramics and carbon-carbon fibers, it becomes clear that the radiative heat is high enough to pose damage to the structural and major stress handling elements within the craft's parameters. The best choice for adopting the best thermal protection equipment could have been the ultra-high temperature ceramics but due to their inability to poses high structural toughness cause us to reject it. It can be observed from the given data that the ultra-high temperature ceramics do have resistance to high ambient heat loads, however, these groups of advanced engineering materials lack another very important characteristic and that is mechanical toughness and crack resistant properties whilst operation. These materials are good to use in static conditions but are proven to be unreliable when these are exposed to hostile interstellar/advanced astronautical missions like reentry in

Earth or foreign planet atmospheres. The lack of mechanical strength in these materials also affect the ability of the material to perform well in high shock environments. However, still it was found that adding a few traces of carbon fibers in to the pre-furnace semi-liquid ceramic matrix that enhances the structural integrity of the ceramic material. Moreover, it was found in arc jet testing facilities that while in operation, these materials were exposed to ambient harsh high shock environment where they reacted harshly with other carbides and nitrides to form different refractory complexes, hence affecting the overall performance of the ceramic material tile.

The best material that can be used are, however, the materials from the group of lightweight ceramic ablators (L.C.A) due to their ease of manufacturing and simplicity of usage. From the arc-jet testing facility based in Mountain-View California, The Ames Research Center, proves subsequently that phenolic impregnated carbon ablators that are formed by impregnating thin and micro-sized carbon fibers with an organic phenolic resin can help the re-entry probe or the launch vehicle to effectively fight the searing high increasing entropy environments with little to no damage.

Extensive research is being conducted in the field of thermal protection systems (T.P.S) in the Ames Research Center in California and world-wide to improve the quality of ablative heat shielding method that involves pyrolyzing the base substance to facilitate mass heat transfer. Various versions of phenolic impregnated carbon ablators.

For example, the Heritage Entry Grade Carbon Phenolic is currently most trusted, capable and robust form of thermal protection system that will be used. The future planned use of the Heritage Entry Grade Carbon Phenolics is mainly for discovery missions far away like Jupiter, Saturn, Uranus or even Neptune.

There are two types of Carbon Phenolics of the Heritage Grade Entry Type: the first one is the tape type and the other one is the chopped type. The tape mask type is used for nose capsules and the chopped type is used for the then re-entry NASA mission types only.

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