Effect of High Temperature Supersonic Flow Conditions over Thermal Barrier Coating over a Flat Plate

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Abstract: For any supersonic missile, one of the most difficult task is to protect the missile's payload and internal components from the excessive heat generated during its flight due to the aerodynamic heating on missile surface. This can be done by applying a thermal barrier coating over missile surface. The performance of any supersonic missile coating can be judged by determining its erosion resistance and the amount of temperature reduction it can give in its substrate. Thus similar conditions were simulated on ground at Scramjet Test Facility in DRDL, Hyderabad. The coating was tested at extreme temperature condition for studying its performance. The shear stress acting at the plate centre was calculated to determine the erosion resistance of the coating. For determining the temperature performance, both bare plate as well as coated plate were tested at flight conditions. The temperature at their` bottom surfaces were measured by using three K type thermocouples. These temperature data were compared to determine the amount of temperature reduction given by thermal barrier coating. After conducting the tests, the thermal barrier coating was found to be efficient for the application on missile surface. This missile coating can protect missile's payload and its internal component thereby increasing missile's efficiency and life.

Keywords: Temperature, Thermal Barrier Coating, Flat Plate, DRDL, Supersonic Flow Conditions.

INTRODUCTION

For supersonic vehicle, theheat transfergenerally occurs at the surface because of aerodynamicviscositywhich ensures that the flow remains at zero speed relative to the body for a very thin layer of molecules at the surface. When fluid flow slows down itskinetic energygets converted into heat. In high speed flows, tremendous amount of heat is generated because of large kinetic energy of the fluid. When the flow is slowed to near zero speed, its kinetic energy gets converted into heat as a result of which the temperature increases and the gradient in the speed in a direction normal to the surface allows small scale mass transport effects to dissipate the temperature in the outward direction and thus the temperature at the surface is less than thestagnation temperature. These viscous dissipative effects to neighboring sub-layers make theboundary layerslow down via a non-isentropicprocess. Heat then conducts into the vehicle surface from the higher temperature air around it. The result is an increase in the temperature of vehicle surface and a loss of energy from the flow. Aerodynamic heating increases with the speed of the vehicle. It produces much less heating atsubsonic speedsbut becomes more important atsupersonic speeds.

When an object travels through air different factors such as skin friction, drag, shear force, viscosity of air etc. acts on the body. In the case of aircrafts, space shuttle and other objects such as missiles when these vehicles travel through fluids such as air, the kinetic energy of air gets converted into heat or thermal energy. This thermal energy is high in the cases of very high velocities. Thus the temperatures over a missile or any object travelling at a velocity of sound can produce a temperature of about 1600K and more than that if we further increase the velocity. Thus this has become a major factor for limitation of speed in missiles, as the structural material cannot withstand that much amount of heat. In order to decrease this effect different methods are used among them TBC can effectively reduce the temperature.

For long range ballistic trajectories one of the most important and difficult phase of flight that designer must cope up with is the re-entry into the earth's atmosphere, wherein the aerodynamic heating associated with the high flight speed is intense. The air temperature in the boundary layer may reach values in the tens of thousands of degrees Fahrenheit which, combined with the high surface shear, promotes very great convective heat transfer to the surface. Heat-absorbent material or thermal barrier coatings must be provided to prevent destruction of the essential elements of missile. The surface shear plays an important role in the convective heat transfer to the surface therefore Reynold's number analogy can be used for the determination of skin friction coefficient and shear stress acting on the missile surface. However, the supersonic convective flow over missile surface can be considered as flow over a flat plate for the determination of skin friction coefficient and shear stress.

LITERATURE REVIEW

Thermal Barrier Coatings are designed to protect metal structural components from extreme elevated temperatures, thereby reducing stress and fatigue and increasing the part's lifespan. In order to provide such a high level of protection, thermal barriers incorporate several key components. Every coating consists of four distinct layers, with each layer adding to the protective thermal properties and enabling the coating to form a unique thermal barrier.

Thermal barrier coatings typically consist of four layers: the metal substrate, metallic bond coat, thermally grown oxide, and ceramic topcoat. The ceramic topcoat is typically composed ofyttria-stabilized zirconia(YSZ) which is desirable for having very low conductivity while remaining stable at nominal operating temperatures

The application of thermal barrier coatings (TBCs) to missiles has facilitated the increase in performance and efficiency of missiles. The thermal barrier coatings can lower the temperature of missile surface which is exposed to very high temperature due to aerodynamic heating. TBCs are essentially multi-layered coating structures that consist of a bond coat that is applied to the substrate, a thermally grown oxide (TGO) and a ceramic topcoat, typically an yttria stabilized zirconia. The bond coat can be applied by either thermal spraying or by diffusion methods. The TGO is formed between the bond coat and the ceramic top coat, during the production of the coatings as a result of the oxidation of the bond coat and continues to grow throughout the life of the coating. Currently there are two basic production routes that are used for modern TBC systems; plasma spraying (PS) – in vacuum for the bond coat and air for the top coat or electron beam (EB) physical vapour deposition (PVD). However, TBCs are susceptible to various life limiting issues associated with their operating environment including, erosion, corrosion, oxidation, sintering and foreign object damage. This research is an experimental project and fully concentrated on the erosion resistance of TBC and the temperature reduction by TBC coating in its substrate.

The main objective was to generate high temperature supersonic flow and carry out the following tests on Thermal Barrier Coating:

1. Temperature reduction given by coating in its substrate.
2. Erosion test.

Conditions given to us: Stagnation temperature = 1500 K, Exit pressure = 1.21 bar, Mach No. at nozzle exit = 1.8

TEST SETUP FOR THE TESTING OF THERMAL BARRIER COATING

4.1 Test plan:-The test was conducted at Scramjet Test Facility (STF) in DRDL, Hyderabad. The subscale vitiated air heater utilized for conducting erosion tests. The setup (Fig.4.1) consists of a Vitiated Air Heater (VAH), a circular to rectangular transition duct (CRTD), 2-D nozzle (facility nozzle) for Mach number 1.8 and the specimen plate holder. The vitiated air heater burns hydrogen, oxygen and air to generate the temperature conditions for testing. Oxygen replenishment was not done for present testing and high temperature conditions were generated burning hydrogen and air only. Air and hydrogen are fed into the heater through a system of high pressure storage cylinders, pressure regulating valve, ON/OFF ball valves and turbine flow meter. The mass flow rate of hydrogen and air to be fed into the heater is determined. Mixture of these gases is ignited in the heater with the help of spark plug. The setup installed at Scramjet Test Facility (STF) is shown in fig. 4.3. The block diagram of the Scramjet
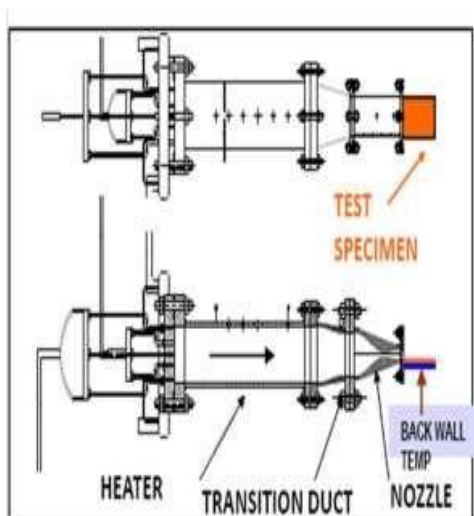Test facility is shown in figure 4.2
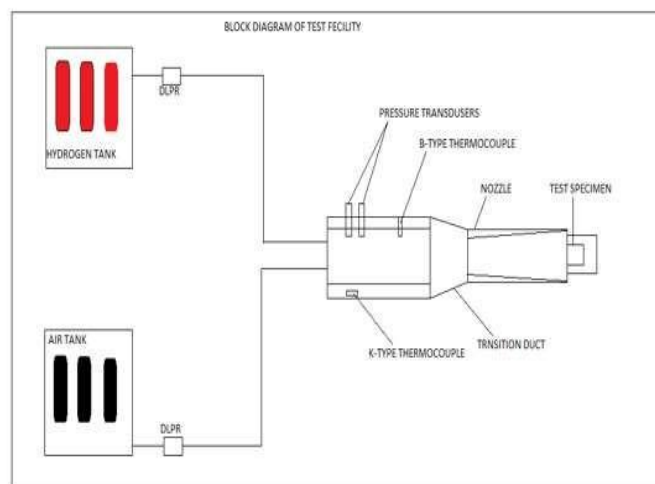

Fig.4.1   Test setup for conducting erosion test   Fig.4.2. Scramjet Test Facility block diagram

Hydrogen bank: -The Hydrogen cylinders are integrated to the master pipe through pipes and hoses. Handling of the hydrogen tanks is very important since hydrogen is highly flammable.

Air bank: - Air Bank consists of the pressurized air stored at 200psi. The dome loaded pressure regulator (DLPR) regulates the air at 200psi and maintains the amount of air entering into the Heater. The reason for using compressed air is to maintain the required mass flow rate in the heater.

Dome Loaded Pressure Regulator (DLPR):- A Dome loaded pressure regulator basically consists of Dome which has a considerable pressure in it. The DLPR is used to regulate the back pressure in the most of the cases. In our case we are using DLPR to regulate the pressure of air and hydrogen which flows into the heater

Turbine flow meter: - The purpose of turbine flow meter is to measure the velocity of the fluid (in our case it's the vitiated air coming out of the heater). The turbine flow meters consists of a turbine upstream to the flow, when the fluid hits the turbine it rotates with very high RPM. This rotational motion is measured and related to the flow velocity.

Heater: - The purpose of heater is to increase the temperature and velocity of the flow in order to simulate the conditions of missile in the desired altitude. In our case the conditions of air coming out of the heater should have: Temperature: 1500 K and Mass flow rate: 1.51 Kg/sec

Challenges faced to generate mass flow rate of 1.51 Kg/sec at 1500 K:-
- The material has to withstand very temperature of 1500 K.
- The time for the combustion of fuel is less.
- There is no extra chamber to mix the fuel, thus the fuel has to be mixed in the combustion chamber itself.
- The time required for mixing of the fuel for keeping it up to the desired stoichiometric point is less (stoichiometric point: it is the point which gives the ideal proportion of the fuel/air mixture for optimum combustion).
- Cooling mechanism must be used to maintain the optimum temperature of the heater.
- Overcoming the challenges faced:-
- Stainless Steel is used to withstand the temperature of 1500 K and very high amount of pressure in the heater.
- In our case hydrogen is used as the fuel since hydrogen has an autoignition temperature of 536°C. Thus, using gaseous fuel over the liquid decreases the time for combustion and also decreases the carbon formation. The air coming out of the heater is a vitiated air since hydrogen also combines with oxygen in air to produce considerable amount of water molecules and other impurities.
- The heater is covered by water jackets in which continuous flow of water takes place thereby, reducing the amount of heat in the heater.

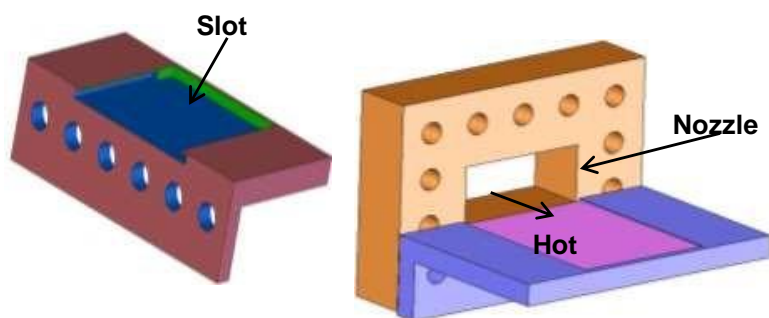Fig.4.3. Test setup at Scramjet Test Facility   Fig.4.4. Specimen holder

The circular cross section of heater is converted into rectangular cross section using a circular to rectangular transition duct. A 2D nozzle is integrated with the heater and transition duct to produce a vitiated hot flow of Mach number 1.8. The nozzle exit cross section is 85mm x 40mm. Specimen plate holder (Fig.4.4) is integrated at the exit of facility nozzle. The plate holder is designed to hold the specimen in place during the test. Coated specimen has to be slid in the slot and the holder has to be attached to the nozzle exit of air heater by means of fasteners.

4.2   Specimen details:-To study the erosion resistance of the LDAM, specimen in the form of a flat plate of dimension 100mm × 120mm × 3mm of Mild Steel was coated with LDAM. The details of specimen is mentioned in table 4.1.

Specimen No.	Coating Material	Coating Thickness(mm)
1.	Bare M.S. Plate	--
2.	TBC	2

The specimen was fitted in flush with the nozzle exit of the air heater and airflow over the coating was tangential during the test 1 and 2, along with tangential flow the coating leading edges were also exposed to flow. These specimens were spot welded with 3 'K' type thermocouple to measure the back wall temperature of specimen during tests as shown in fig. 12.1

4.3 **Measurement plan:-**Strain gage based pressure sensors for static pressure measurement, type 'B' temperature sensors for flow stagnation temperature and type 'K' sensors for outside surface temperature measurement, were used. Turbine flow meters with flow computers were used to measure mass flow rates of individual gases. The sensors used for measurements were calibrated and accuracy found to be within acceptable limits. NI - DAQ Lab view and SCADA system were used to acquire the pressure, temperature and flow data. TBC plate holders are designed to hold the coated specimen to the nozzle exit of the air heater

Table 4.2.Pressure measurement plan

S. No.	Nomenclature	Location
1	VAHP	Vitiated Air Heater pressure
2	NOZ_EXIT	Exit of nozzle, static pressure
3	H2_INJ	Hydrogen Injection Pressure
4	AIR_INJ	Air Injection Pressure

Table 4.3.Temperature measurement plan

S.No	Nomenclature	Location
1	SMat01	Type 'K' thermocouple, spot welded at the back wall of the specimen (Fig. 3)
2	SMat02	Type 'K' thermocouple, spot welded at the back wall of the specimen (Fig. 3)
3	SMat03	Type 'K' thermocouple, spot welded at the back wall of the specimen (Fig. 3)

Table 4.4.Flow measurement plan

S. No.	Nomenclature	Location
1.	Air flow rate	Air feed system
2.	Hydrogen flow rate	Hydrogen feed system

Reason for using K type thermocouple to measure surface temperature and B type thermocouple to measure gas flow temperature in the heater:-Different types of thermocouples have different ranges of operation for an example a 'B' type thermocouple has a rage of 870° C to 1700° C whereas a 'K' type thermocouple has a range from 293° C to 1260° C. Since the temperature of the skin will be less in comparison with the gas flow temperature therefore, different thermocouples are used for the measurement purpose. Different types of thermocouples and their wide ranges of operation are shown in figure 4.6.

Fig.4.6. Tolerance of thermocouples

4.4 Firing sequence:- The sequence for starting and ending of different feed system and data recording is shown in table 4.5. Also the feed system duration is shown in fig.4.7. From fig.4.7 and table 4.5, it is clear that the hot flow of air is started at t=0 (Spark Ignition on) and stopped at t=16s (Heater hydrogen off). Thus the test was aimed for 16s. Table 4.5.Typical firing sequence

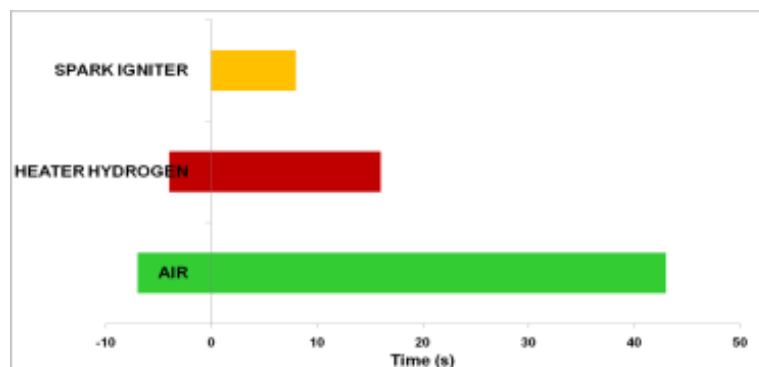| S. No.	Event	Time (sec)
1	DATA ACQUISATION ON	-20
2	AIR ON	-7
3	HEATER HYDROGEN ON	-4
4.	SPARK IGNITION ON	0
5	SPARK IGNITION OFF	8
6	HEATER HYDROEGN OFF	16
7	AIR OFF	43
8	DATA ACQUISATION OFF	48

Fig.4.7.Feed system timing during test

4.5 **Typical pressure data:-**Fig. 4.8(a) shows the heater and nozzle exit static pressure during the test whereas the fig. 4.8(b) shows the typical air and hydrogen injection pressures. The hot flow is indicated in the figure, which is the same as the test duration
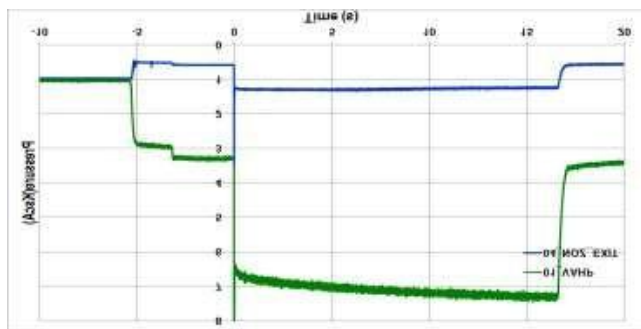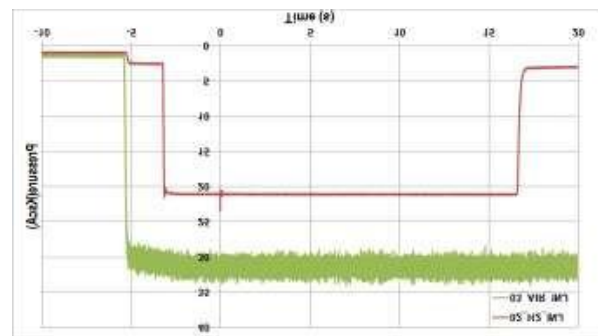

Fig.4.8 (a).Heater and nozzle exit pressure   Fig.4.8 (b).Air and hydrogen injection pressure

ASSUMPTIONS

1. The nozzle is considered to be a Divergent nozzle with constant change in the area (keeping θ as constant). The area change is assumed to be constant for the simplification of our calculation.
2. The specific heat constant is taken as constant ($\gamma = 1.3$). The specific heat constant in the nozzle will keep on changing with the change in temperature. The C_P and $C_V$both increases along the nozzle but the specific heat constant (γ) decreases as the temperature decreases due to the expansion of gases in the nozzle. Calculation of specific heat constant at every segment is very difficult thus, the specific heat constant is assumed to be a constant
3. The flow in the heater and nozzle is considered to be isentropic flow. Since it is complicated to calculate the various flow relations for an entire flow, a constant volume is considered. It is also been considered that the temperature is not added or escaped thus being an adiabatic and reversible process. Hence isentropic flow is assumed for the calculation of various flow parameters.

4. Exit pressure is taken as 1.2 bar to prevent the boundary layer separation and to maintain a smooth exit flow. Since the simulation is done on the ground level, the pressure at the ground level where the test has been conducted is 1.2 bar. 1.2 bar exit pressure is taken into consideration so as to prevent certain losses such as boundary layer separation and to prevent over expansion condition. The exit pressure is maintained to be equal with the ground level pressure

5. The frictional losses in the nozzle are neglected. For the project, our aim is to produce supersonic flow at Mach No. 1.8 thus, the frictional losses are omitted since they are very less.
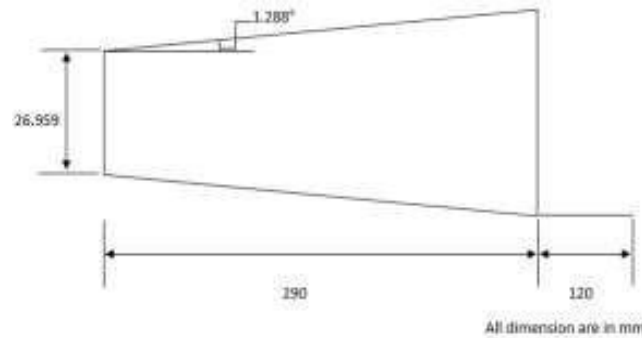
SOLVING METHODOLOGY

Fig.6.1.Simplified setup considered for analysis

1. The diverging part of the nozzle is divided into segments of 10 mm length as shown in Fig.6.1.
2. Mach number for various stations along the length of nozzle is obtained from area ratio by using area variation of nozzle.
3. Local values of static temperatures are calculated by taking stagnation temperature measured in VAH and local values of ratio of specific heats obtained from area ratio.
4. Local velocities are calculated as, $_{local} = M_{local} \times (\gamma.R.T)^{0.5}$

Local value of Gas constant 'R' is used for calculation. The value of R = $\frac{8314}{26.6}$ = 312.556 J/Kg K

5. Using local values of static temperatures and velocities, average values for the segments are calculated.
6. Average Mach numbers for the segments are calculated using average static temperatures and velocities.
7. Average values of densities for the segments are calculated in a similar way. $^T1.5$
8. Viscosity is calculated using Sutherland's Law considering average T_{static}. $\mu = 1.46 \times 10^{-6} (\underline{\quad\quad}) T + 111$
9. Reynolds number for each segment is calculated considering average values of density, velocity and viscosity for the segment. The characteristic dimension for each segment is the segment length of 10 mm.
10. Cumulative Reynolds number at the exit of nozzle, $_{enozzleexit} = R_{e1} + R_{e2} + R_{e3} + \ldots\ldots\ldots\ldots\ldots\ldots + R_{e29}$
11. Reynolds number at the centre of the plate is calculated as follows: $R_{eplatecentre} = \rho \underline{\quad\quad\quad\quad\quad\quad\quad\quad}$ $nozzleexit_\mu nozzle \times Vnozzleexitexit \times l/2 + Renozzleexit$
12. The nozzle exit Mach number is 1.8. Skin friction coefficient (C_{fx}) for adiabatic wall is obtained from heat and mass transfer data book, using nozzle exit Mach number and Reynolds number at the centre of the plate.
13. Shear Stress at plate centre is calculated as follows: $\tau = C_{fx} \times 1/2 \times \rho_{nozzleexit} \times V^2_{nozzleexit}$
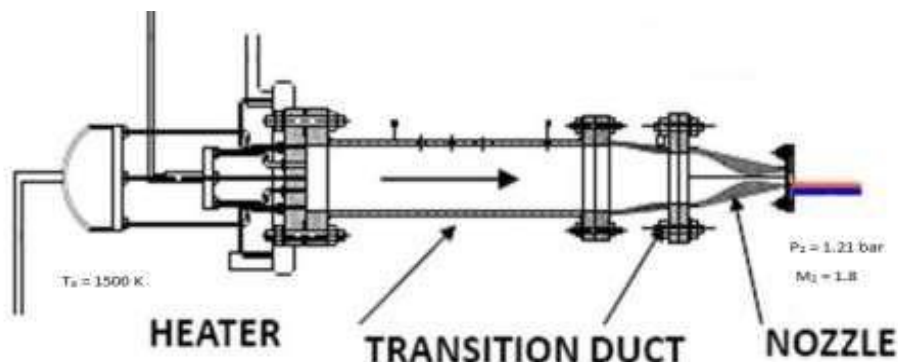
CALCULATION OF FLOW PARAMETERS

Fig.7.1.Test setup with various conditions

The test conditions are as follows:-

Total temperature (T_o) = 1500 K, Exit Mach No. (M_2) = 1.8, Pressure at nozzle exit (P_2) = 1.21bar,
Specific heat constant (γ) = 1.3, From Temperature-Mach no. relation, $\frac{T_O}{T_2} = 1+M_2^2\frac{1}{2}(\gamma-21)$ Where, T_2 = Temperature at nozzle exit, $\frac{1500}{T_2} = 1+1.8^2\left(\frac{1.3-1}{2}\right)$, T_2 = 1009.5 K.

From pressure-Temperature relation, $\frac{P_O}{P_2} = \left(\frac{T_O}{T_2}\right)^{\gamma^{-1}}$, Where, P_O= Total pressure, so $\frac{P_O}{1.21} = ()$, $P_O = \left(\frac{1009.5}{1500}\right)^{\frac{1.3}{0.3}\gamma}$ 6.773 bar

Now, by using gas tables:- At M_2 = 1.8 and γ = 1.3, A_2/A^* = 1.484
Where, A_2 = Area of nozzle exit, A^* = Area of throat, Area of throat (A_2) = 85*40 = 3400 mm^2

Therefore, by using area ratio, $\frac{3400}{A^*} = 1.484$, A^* = 2291.105 mm^2
Now, A^* = width * Height, Here, Width = 85 mm, On substitution, 2291.105 = 85 * h, h =26.954mm.

$$(m) = \frac{A}{T_O} \sqrt{R} \, M_2 \left(1 + M_2 \frac{}{2}\right)^{-(\gamma+1)}$$

Now, the mass flow at nozzle exit $\frac{P_O}{\sqrt{}} \gamma$ $2^{\gamma-1} 2(\gamma-1)$ =1.5122kg/sec. (M=1 under chocked condition).

TABULATION OF VALUES

The values at every section are calculated and then tabulated as follows:-

Distance from the throat (mm)	Area ratio	Mach No.	$\frac{T}{T_0}$	Local Static Temperature, T (°C)	$\frac{P}{P_0}$	Local Static Pressure, P (bar)
10	1.0166	1.1424	0.8362	1254.40	0.4607	3.120
20	1.0327	1.2019	0.8219	1232.85	0.4274	2.894
30	1.0455	1.2394	0.8127	1219.05	0.4071	2.757
40	1.0625	1.2822	0.8022	1203.30	0.3847	2.605
50	1.0795	1.3196	0.7929	1189.35	0.3658	2.477
60	1.0965	1.3533	0.7845	1176.75	0.3493	2.365
70	1.1135	1.3842	0.7767	1165.05	0.3346	2.266
80	1.1305	1.4128	0.7695	1154.25	0.3214	2.176
90	1.1475	1.4395	0.7628	1144.20	0.3094	2.095
100	1.1645	1.4650	0.7564	1134.60	0.2983	2.020
110	1.1835	1.4918	0.7497	1124.55	0.2870	1.943
120	1.2001	1.5141	0.7441	1116.15	0.2778	1.881
130	1.2168	1.5355	0.7387	1108.05	0.2692	1.823
140	1.2335	1.5562	0.7335	1100.25	0.2611	1.768
150	1.2502	1.5760	0.7285	1092.75	0.2535	1.716
160	1.2669	1.5951	0.7237	1085.55	0.2463	1.668
170	1.2836	1.6136	0.7191	1078.65	0.2396	1.622
180	1.3002	1.6314	0.7147	1072.05	0.2332	1.579
190	1.3169	1.6488	0.7103	1065.45	0.2271	1.538
200	1.3336	1.6657	0.7061	1059.15	0.2214	1.499
210	1.3503	1.6821	0.7020	1053.00	0.2159	1.462
220	1.3670	1.6981	0.6980	1047.00	0.2106	1.426
230	1.3837	1.7137	0.6942	1041.30	0.2056	1.392
240	1.4004	1.7289	0.6904	1035.60	0.2008	1.360
250	1.4171	1.7437	0.6867	1030.05	0.1963	1.329
260	1.4337	1.7581	0.6832	1024.80	0.1919	1.299
270	1.4504	1.7723	0.6797	1019.55	0.1877	1.271
280	1.4671	1.7862	0.6763	1014.45	0.1837	1.224
290	1.4837	1.7997	0.6730	1009.50	0.1798	1.217

Distance from the throat (mm)	Dynamic Viscosity,μ (Kg/m sec)	Local velocity, V (m/sec)	Local density,ρ (Kg/m^3)	Reynold's Number, R_e
10	4.7506×10^{-5}	815.5914	0.7957	136607.181
20	4.7029×10^{-5}	850.6659	0.7510	135847.160
30	4.6721×10^{-5}	872.2835	0.7235	135077.872
40	4.6351×10^{-5}	896.5578	0.6926	133973.987
50	4.6045×10^{-5}	917.3450	0.6663	132749.553
60	4.5766×10^{-5}	935.7757	0.6430	131476.015
70	4.5499×10^{-5}	952.3722	0.6222	130253.889
80	4.5251×10^{-5}	967.5339	0.6031	128962.469
90	4.5018×10^{-5}	981.5179	0.5858	127720.730
100	4.4795×10^{-5}	994.7057	0.5696	126481.017
110	4.4562×10^{-5}	1008.4060	0.5527	125072.033
120	4.4365×10^{-5}	1019.6507	0.5391	123902.556
130	4.4174×10^{-5}	1030.3033	0.5263	122771.551
140	4.3990×10^{-5}	1040.5110	0.5141	121601.888
150	4.3812×10^{-5}	1050.1521	0.5024	120422.810
160	4.3641×10^{-5}	1059.3718	0.4916	119334.382
170	4.3476×10^{-5}	1068.2470	0.4811	118210.882
180	4.3318×10^{-5}	1076.7219	0.4712	117122.526
190	4.3159×10^{-5}	1082.2191	0.4618	115797.117
200	4.3008×10^{-5}	1092.7255	0.4528	115045.132
210	4.2859×10^{-5}	1100.2758	0.4442	114034.977
220	4.2713×10^{-5}	1107.5725	0.4357	112979.500
230	4.2574×10^{-5}	1114.7008	0.4276	111957.077
240	4.2435×10^{-5}	1121.5057	0.4201	111043.203
250	4.2299×10^{-5}	1128.0712	0.4127	110062.881
260	4.2170×10^{-5}	1134.4849	0.4055	109090.260
270	4.2041×10^{-5}	1140.7148	0.3988	108207.954
280	4.1915×10^{-5}	1146.7823	0.3860	105616.693
290	4.1793×10^{-5}	1152.6270	0.3857	106373.850

Table 9.1 (a).Various flow parameters at each nozzle segment n   Table. 9.1 (b). Various flow parameters at each nozzle segment

VARIATION OF PROPERTIES ALONG THE NOZZLE
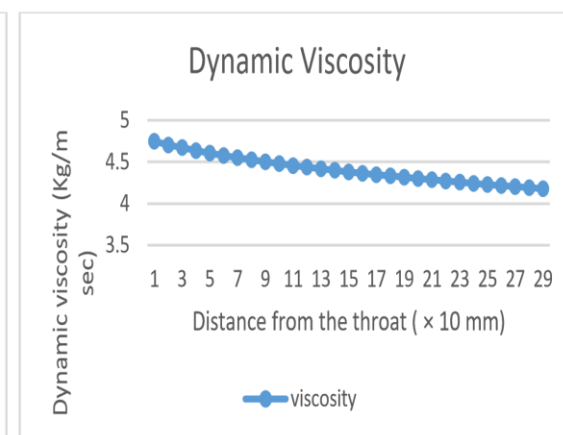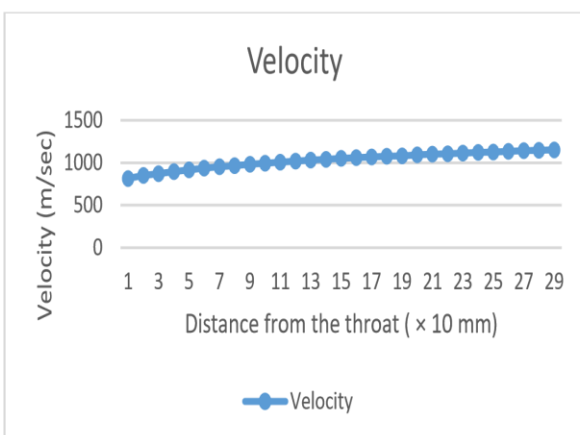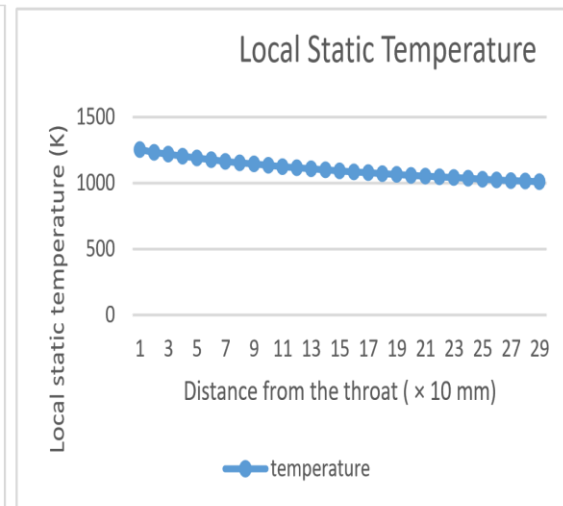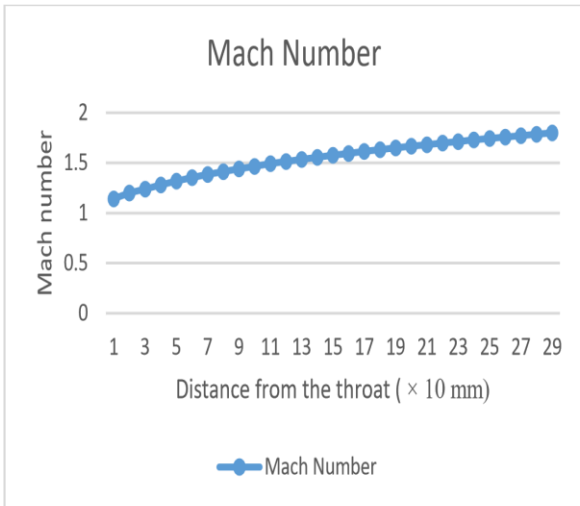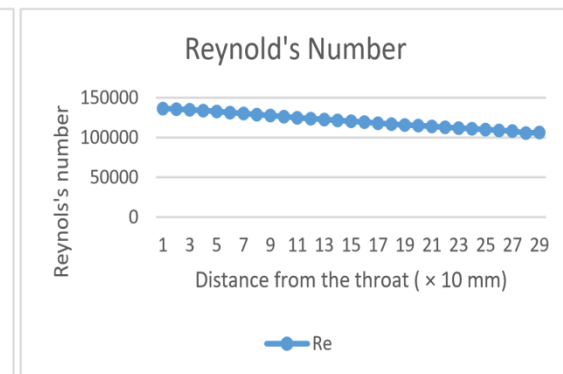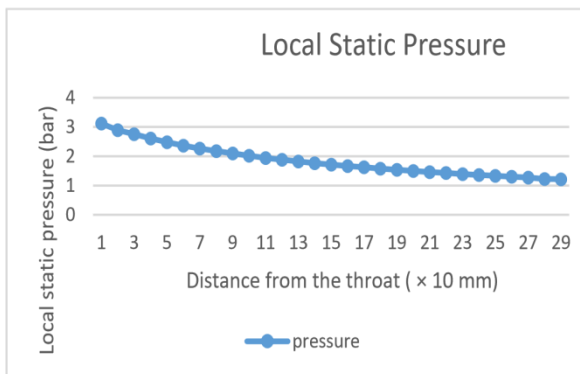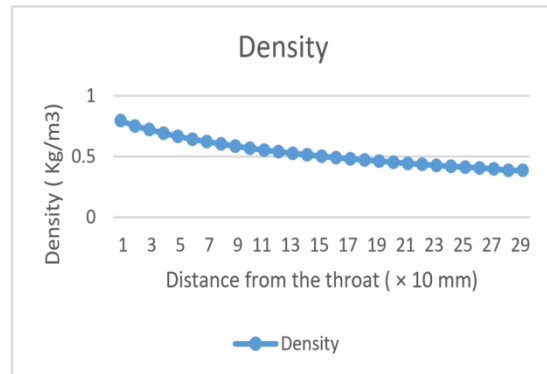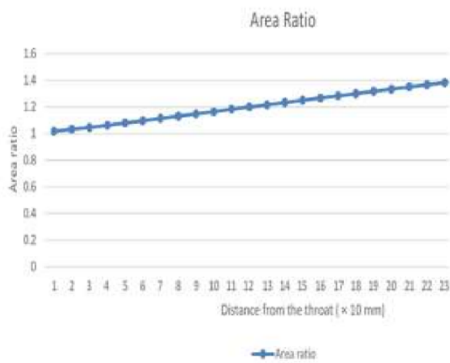
SHEAR STRESS CALCULATION

The Reynold's number at nozzle exit is given by,

Renozzle exit = Re1 + Re2 + Re3+.. . . . . +Re29

By using the Reynold's number values at each segment, $R_{nozzle\ exit}$ = 3507797.147

Now, The Reynold's number at plate centre is given by, $R_{plate\ centre}$ = ρ_____nozzle exit$_\mu$×Vnozzle exit×$1/2$ + Renozzle exit

nozzle exit

Where, $\rho_{nozzle\ exit}$= Density at nozzle exit = 0.3875 Kg/m^3 $V_{nozzle\ exit}$ = Velocity at nozzle exit = 1152.627 m/sec $\mu_{nozzle\ exit}$ = Dynamic viscosity at nozzle exit = 4.1793 × 10^{-5} Kg/m sec,   thus, $R_{plate\ centre}$ = 4.1490 × 10^6

Calculation of skin friction coefficient(C_{fx}):- C_{fx}= 0.0591$R_{ex}^{-0.2}$ {5 × 10^5 < R_e < 10^7} = 2.81017 × 10^{-3}

[From Heat and mass transfer data book by S.M Yahya]

Calculation of shear stress (τ):- $\tau = C_{fx} \times 1/2 \times \rho_{nozzle\ exit} \times V^2_{nozzle\ exit}$= 719.9955 N/m^2

FINDING TEMPERATURE PERFORMANCE OF THERMAL BARRIER COATING

In order to check the performance of the thermal coating the temperature of the bare plate in the supersonic flow is calculated and compared with the plate coated with TBC.

12.1 Tests conducted: - Test No. 1:-

Specimen   : Mild Steel bare plate,   Size: 120mm x 100mm,   Thickness   : 3mm

- K-type thermocouples are installed at the bottom of the bare M.S. plate as shown in figure 12.1.
- Then this plate is kept in the flow using the plate holder as show in the figure.
- Once the fluid flow is started, the flow passes over the bare M.S. plate at a temperature of 1009.5 K. ☐ The visual conditions of the plate before and after the test is shown in the figure 12.2.
- The temperature at the bottom surface of the plate is measured by using type 'K' thermocouples.
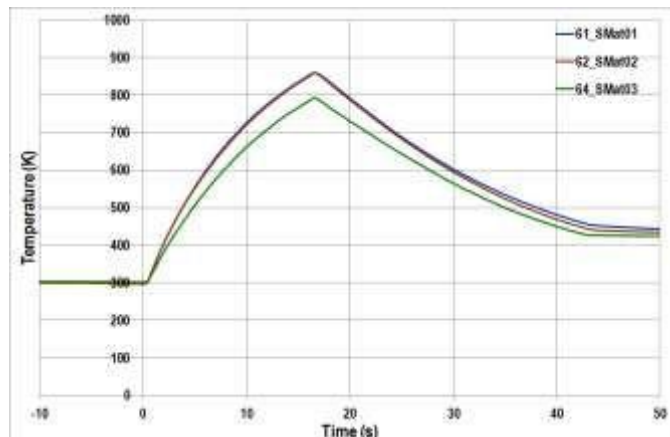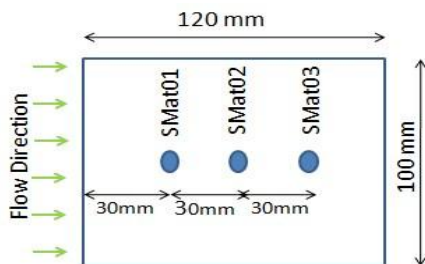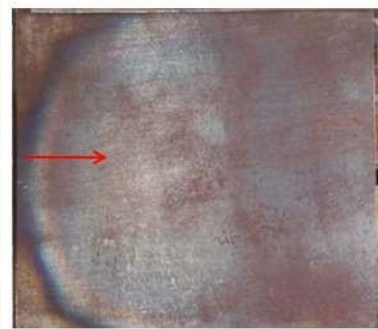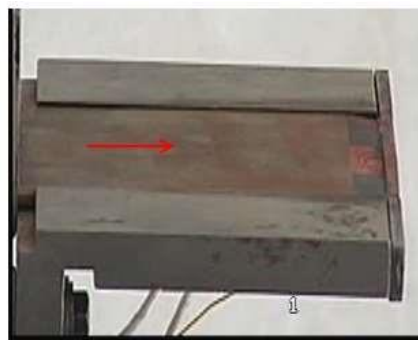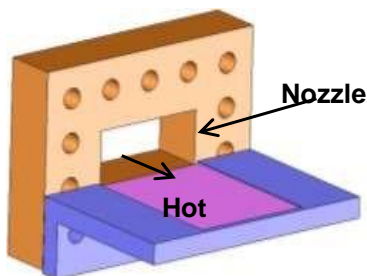

Fig.12.1.Location of type 'K' thermocouples bonded to the plate.   Fig.12.3.Back wall temperature specimen-1,

Mild Steel bare plate, 3mm thickness. Figure 12.3 shows the back wall temperature for the bare Mild Steel plate, 3mm thick. Figure 12.2 (a) and 12.2 (b) show the condition of the specimen before and after the test. The discoloration of the plate is seen in the test.

12.1(a).Mild Steel bare plate before the test   12.2(b).Mild Steel bare plate after the test

Observation test no.1:-

1. Test duration is approximately 16 seconds.
2. Maximum temperature reached is 834.3 K by SMat01 at the end of 16th second.
3. Thermocouples SMat02 are reading almost identical and responding in a similar way. Thermocouple SMat03 is also reading in similar way but temperature recorded is lesser than SMat01 and SMat02.
4. This test will be base for comparing the performance of TBC coating testes subsequently.

Test No. 2:-

Specimen: TBC coating,   Size: 120mm x 100mm,   Thickness: TBC coating, 2mm and M.S. plate substrate 3mm

☐The TBC coating is applied to another test specimen of same material of bare plate and rest all conditions were same as conditions set during bare plate.

Observation test no. 2:-

1. Temperature recorded at the back wall of specimen at the 16th second of test duration by SMat01, SMat02, SMat03 are 327 K, 323 K and 326 K respectively. These all values are approximately equal.
2. Temperature reduction of approximately 500 K at 16th second of operation is measured with respect to bare plate at SMat01 and SMat02.
3. Mild erosion is seen in the coating.
4. Sufficient thickness of TBC material is observed on the specimen after the test.

12.2 Test results: After measuring the temperature values at the bottom side of the plates, the graphs (Temperature vs Time) are plotted as shown in figure 12.3 and figure 12.5. The maximum temperature reached in bare M.S. plate is 834.3 K whereas, the maximum temperature reached in the plate coated with TBC is 327 K. Temperature reduction of approximately 500 K is achieved in the plate coated with TBC as compared to the bare plate.
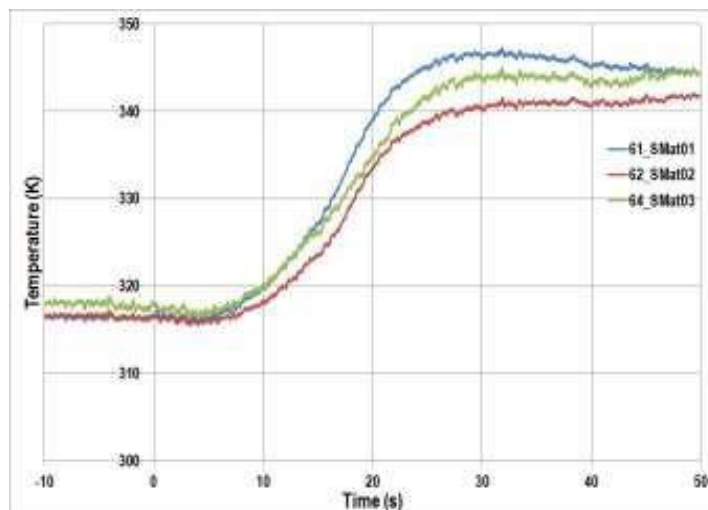

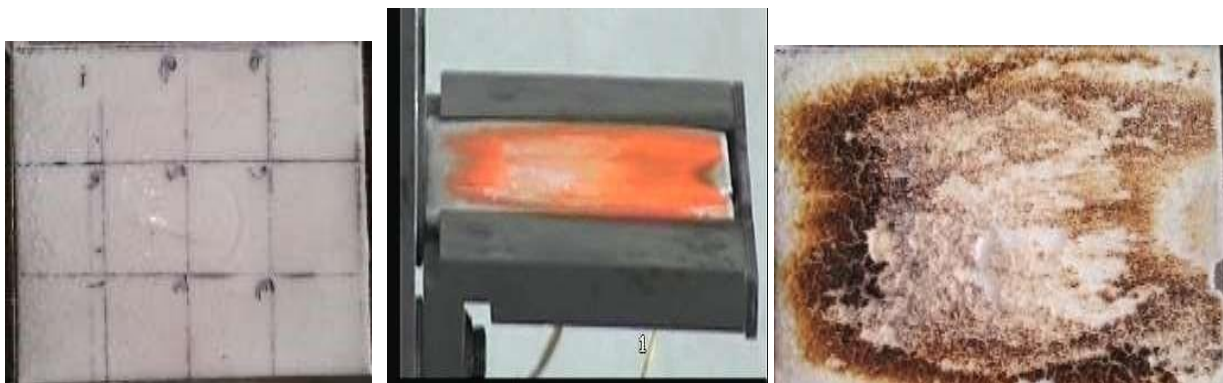Fig.12.5.Back wall temperature specimen-2, TBC coating, 2 mm thickness

Fig.12.4 (a).Before test   Fig.12.4 (b).During test   Fig.12.4 (c).After test

CONCLUSION

The Thermal Barrier Coating was successfully tested at Scram Test Facility under high temperature supersonic flow conditions.

Temperature reduction of approximately 500 K was achieved in the Mild Steel plate coated with TBC as compared to the bare M.S. plate. The Thermal Barrier Coating can successfully withstand maximum shear stress of about 720 N/m^2 at exit flow temperature of 1009.5 K. Sufficient thickness of TBC material was observed on the specimen after the test. The coating did not vanish completely during the 16 seconds of test duration and maintained sufficient thickness after the test. Thus, this missile coating is suitable for the application in supersonic missiles since, a supersonic missile undergoes a wide range of temperature changes during its course. This missile coating can effectively protect missile components and its payload from excessive heat generated due the aerodynamic heating of missile surface while maintaining sufficient thickness for longer period of time.

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