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Experimental Hypersonic Investigation over the Micro-Satellite SARA

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Abstract. Brazilian Space Agency has proposed the development of experiments in a microgravity environment, using a recoverable orbital system. The system consists of a capsule shaped orbital platform. After the microgravity experiments were carried out, the reusable space vehicle will return to Earth and it will be recovered at ground. To understand the hypersonic flow characteristics, considering the aerodynamic aspects during the ballistic re-entry flight, of the returnable space vehicle, a 10-cm diameter Aluminum model has been fabricated and it has been instrumented with piezoelectric pressure transducers and thin-film platinum heat gages. The experiment has been performed in the IEAv 30-cm. diameter Hypersonic Shock Tunnel at freestream Mach from 6 to 8 with stagnation temperature from about 5000K to 950K, which it is capable of producing air flow in the test section up to 5000 m/s, with stagnation enthalpy and pressure up to 7.5 10^6 J/kg (5000 K stagnation temperature) and 150 atm, respectively. The useful test time from 0.5 to 1.5 milliseconds is enough time to obtain the aerodynamic parameters such as pressure and heat flux distributions. Pressure and heat flux distributions over the model and the photographs of the hypersonic flow are presented.

Keywords: *Reusable Space Vehicle, Hypersonic Flow, Blunt Body, Experimental Pressure Measurement, Experimental Heat Transfer Measurement, Hypersonic Shock Tunnel.*

1. Introdução

The primary objective of the present experimental investigation is to provide experimental data required for understanding the hypersonic flow characteristics over a reusable space vehicle. Such vehicle is intended for scientific and technological experiments in low-gravity environment, which it enables the production of homogeneous and perfect chemical crystals, the creation of new metallic alloys, new electronic components, unaccountable agronomic and biomaterial researches (Moraes Jr, 1997)...The design of the micro-satellite (≈ 150 kg) (recoverable at ground and reusable) performing scientific and technological experiment in low-gravity environment ($\approx 10^{-5}$ g), for short time (about 10 days), in equatorial orbit (≈ 300 km), Fig. 1, has been proposed by the Brazilian Space Agency.

Among other applications (Moraes Jr, 1997) of this reusable orbital platform, there are: *i)* experiments in low orbit (micro-gravity, vacuum, direct solar radiation, low temperature); *ii)* studies of re-entry flow and *iii)* as technological demonstrator. The main users are: Universities, research institutes, national industries, national and international space programs…

Returnable space vehicles undergo several velocity regimes and flight conditions that may complicate their aerodynamic design. Only gravitational and drag forces $\left({}^{lift}\!/_{drag} \approx zero \right)$ act during the ballistic re-entry flight of reusable space vehicle into the Earth atmosphere. At altitude $\approx$ 300 km the gravitational field is nearly constant. At altitudes higher than 90 km the drag force is very low. Consequently, the velocity of the <u>A</u>tmospheric <u>R</u>eusable <u>SA</u>tellite (*SARA*) does not vary significantly in altitudes higher than 100 km, Fig.2. The velocity increment, at the beginning of the *SARA* re-entry trajectory, is due to the earth gravity in the rarefied environment. When the *SARA* goes to lower altitudes, its velocity decreases (from 8000 km/s to 80 m/s in about 5 minutes). Consequently, the reduction of velocity results in conversion of kinetic energy into thermal energy, due to the friction, between the microsatellite external surface and the Earth atmosphere (Pessoa-Filho, 1997).

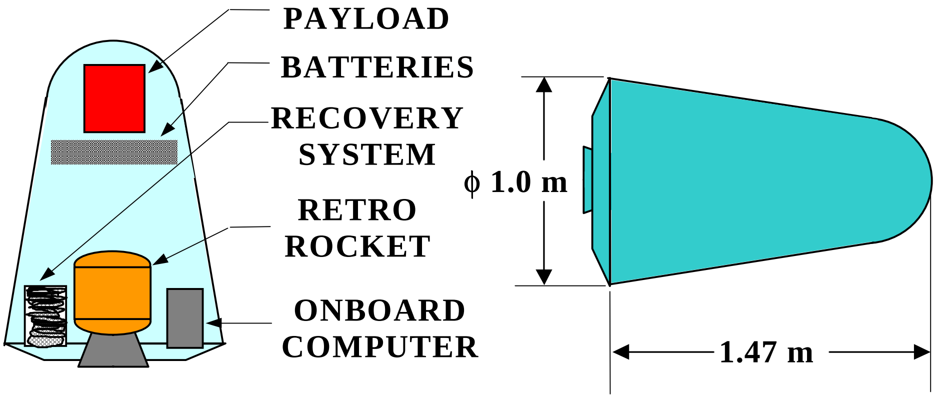


Figure 1: Preliminary concept of the Reusable Space Vehicle.

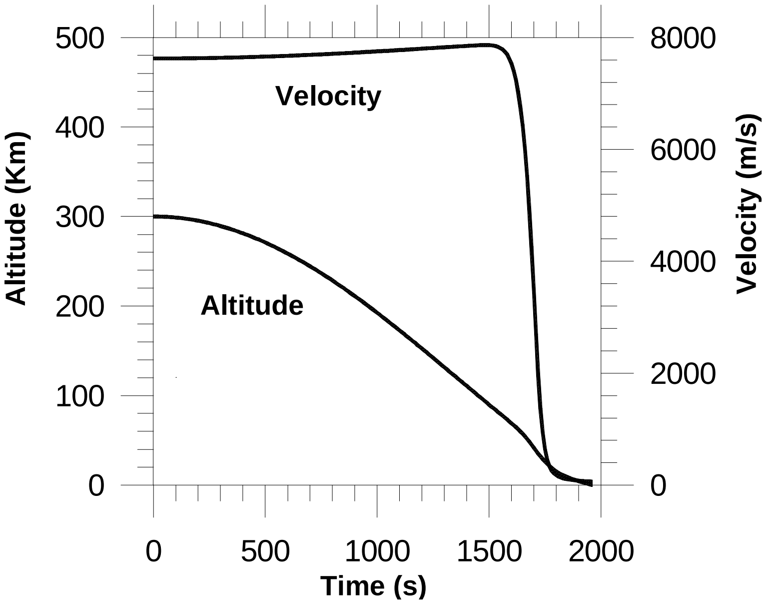


Figure 2: *SARA* re-entry typical trajectory.

The feasibility of transatmospheric flight is limited by phenomena such as aerodynamic drag and heating as well as related thermal management problems. Therefore, an efficient transatmospheric hypersonic vehicle design has to combine a low drag coefficient (to maximize the net propulsive thrust) with low heat transfer rates (to minimize thermal protection system mass).

Traditional blunt-nosed hypersonic vehicles generate a strong detached normal shock wave in the nose region, which produces high aerodynamic drag. The temperature behind this strong shock wave increases at hypersonic velocities, although the aerodynamic heating rates are reduced compared with that of an attached shock wave on a conical body. On the other hand, a traditional slender body with a sharp leading edge produces a conical weak attached shock wave with low drag coefficient, but extreme heating is created at the tip of the forebody.

For take off, escape from and flight through the earth's atmosphere the drag on the body should be reduced. For re-entry vehicle into the earth's atmosphere at hypersonic speeds, it is important to have a large nose radius, consequently, low aerodynamic heating, Fig. 3.

For the drag coefficient case, Fig. 3, C_D = 0.4, the maximum heat flux (at the stagnation point of *SARA* nose) is about 2.5 MW/m^2 during about 5 minutes of re-entry trajectory (Pessoa Filho, 1997). In this condition, the temperature of the air, after the detached normal shock wave is about 6500K (the ambient air temperature is about –30°C). On the other hand, for the highest drag coefficient case, C_D = 1.2, the maximum heat flux (at the stagnation point) is about 0.8 MW/m^2 at the same re-entry trajectory.

Heat flux at stagnation point, of *SARA* nose using Van Driest (1952) method, Fig. 4 (Miranda and Mayall, 2001), agrees with the Pessoa Filho (1997) heat flux results, Fig. 3. The calculations using the Tauber and Menees (1986) method, Fig. 4, are more conservative than those presented by Pessoa Filho (1997) and Van Driest theory.

The goal is to simulate the hypersonic flow (Mach number and stagnation temperature) over the *SARA* configuration (capsule shaped orbital platform), with the lowest drag coefficient, using the IEAv 30-cm. diameter Hypersonic Shock Tunnel at freestream Mach from 6 to 8 with stagnation temperature from 5000K to 950K.

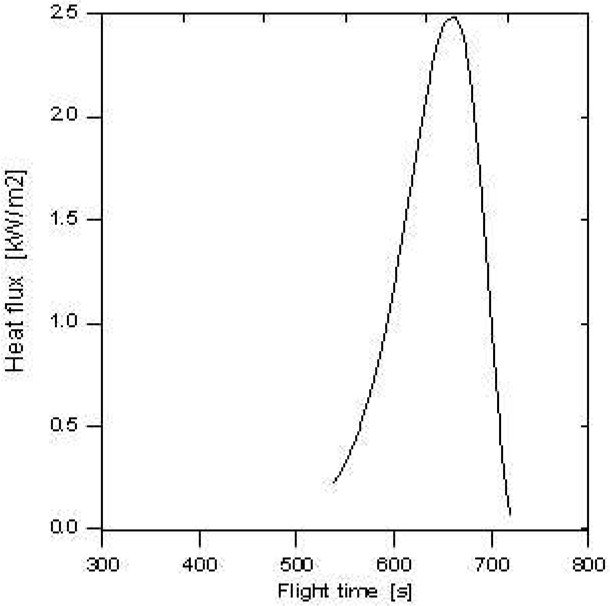
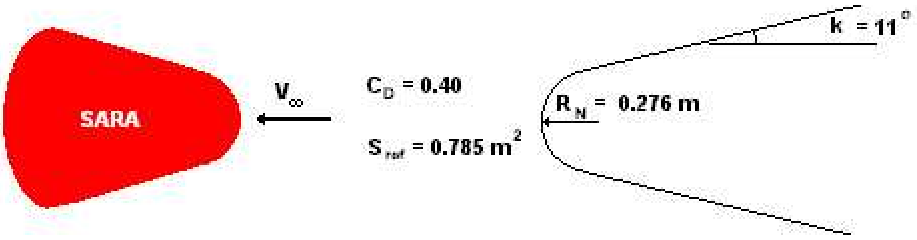


Figure 3: Heat flux for drag coefficient, C_D = 0.4, Pessoa Filho (1997).



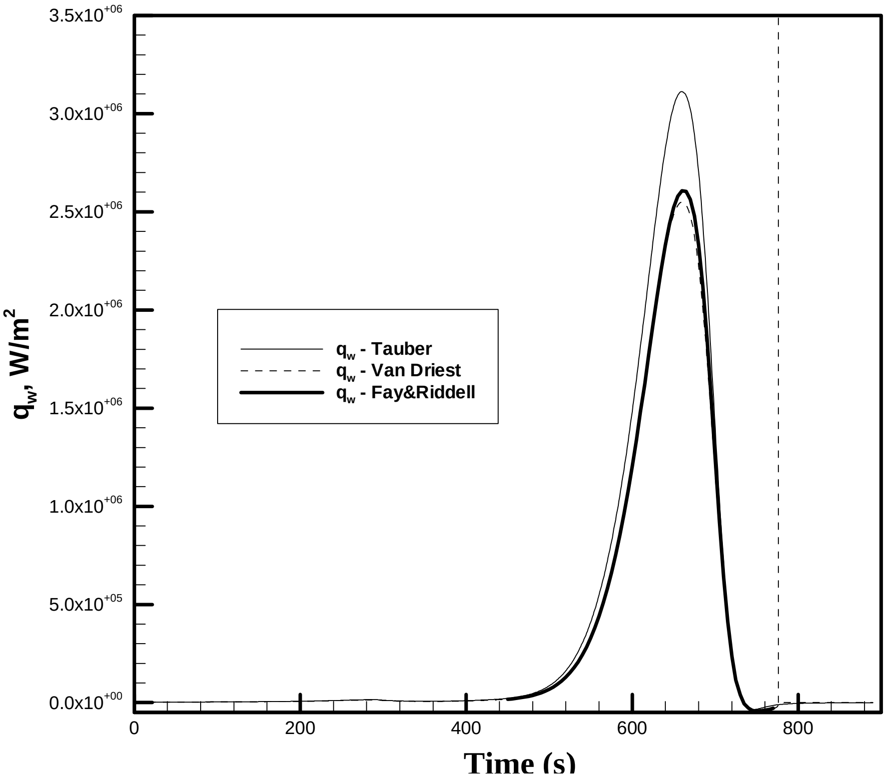


Figure 4: Heat flux for drag coefficient, C_D = 0.4, Miranda and Mayall (2001).

2. IEAv 30-cm diameter Hypersonic Shock Tunnel

Shock tubes and shock tunnels are the most versatile facilities for short duration hypersonic flow experiments (Nagamatsu, 1961). They have been widely used for high velocity and high temperature research since the early 1950s. Both shock tubes and shock tunnels are ground test facilities which provide high enthalpy flows close to those encountered during the reentry of a space vehicle into the earth's atmosphere at hypersonic flight speeds. These short running time facilities require special measurement techniques to measure pressure and temperature (heat flux) for the models in the test section.

The IEAv 30-cm. diameter Hypersonic Shock Tunnel (Minucci et al., 2001), Fig. 5, is capable of producing air flow in the test section up to 5000 m/s, with stagnation enthalpy and pressure up to 7.5 10^6 J/kg (5000 K stagnation temperature) and 150 atm, respectively. The useful test time is about 1.5 milliseconds, enough time to obtain the aerodynamic parameters such as pressure and heat flux distributions.

Figure 5: IEAv 30-cm. Diameter Hypersonic Shock Tunnel.

1. Experimental Apparatus

The model geometry, Fig. 6, consists of a capsule shaped orbital platform, wherein the front nose is a hemispherical shape followed by a conical segment. A 10-cm. diameter aluminum model has been constructed to investigate the incident hypersonic flow in the IEAv 30-cm. diameter Hypersonic Shock Tunnel, Fig. 5. The model was designed to enable the measurement of pressure drag and heat flux across the model forebody at hypersonic speeds.

Figure 6: Ballistic re-entry vehicle *SARA* model.

4. Experimental Results

The nominal shock tunnel test conditions, to run the low drag SARA configuration, Fig. 6, are presented in Table 1. These conditions did not vary more than 5% from run to run.

Time-lapse type photographs of the luminous air flow around the model were taken by using a Nikon camera model N6006 with AF35-70mm f/3.3-f/4.5 Nikkor lenses and ISO 100 color film.

All the data, with the exception of the flow visualization, were recorded using a Tektronix VX4244 16-channel 200kHz data acquisition system.

Table 1. Hypersonic Shock Tunnel Test Conditions

		High Enthalpy	**Medium Enthalpy**	**Low Enthalpy**
Stagnation Conditions	**Pressure**	120 bar	140 bar	20 bar
	Temperature	5.000 K	2.000 K	950 K
	Enthalpy	9 MJ/kg	2 MJ/kg	1 MJ/kg
Freestream Flow Conditions	**Pressure**	12 mbar	4 mbar	4 mbar
	Temperature	1.000 K	192 K	77 K
	Density	4.0 g/m^3	33 g/m^3	17 g/m^3
	Mach	6,2	7,3	7,8

Figures 7 and 8 display typical reservoir and impact pressure traces present for low enthalpy air conditions. The reservoir pressure trace as well as the impact pressure trace show the test time is about 700 μs (the voltage is almost constant).

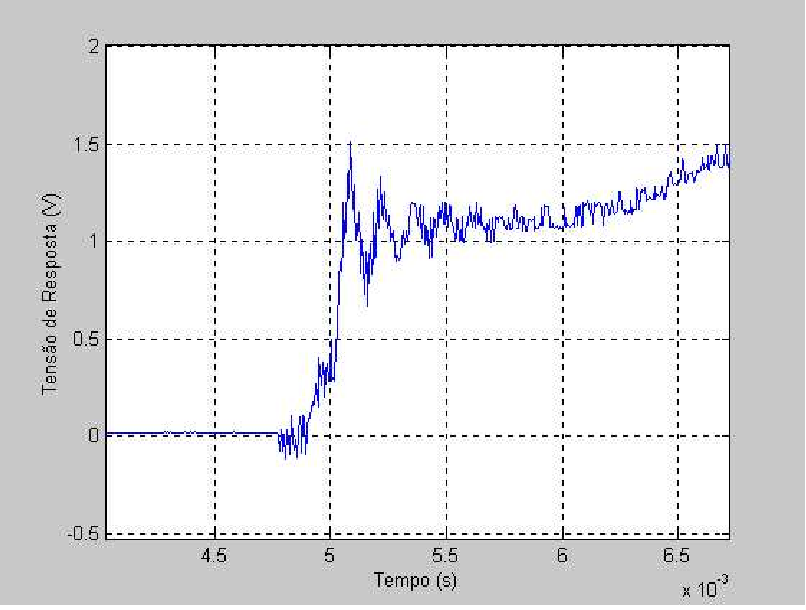


Figure 7: Typical pressure trace (voltage trace) for reservoir pressure.

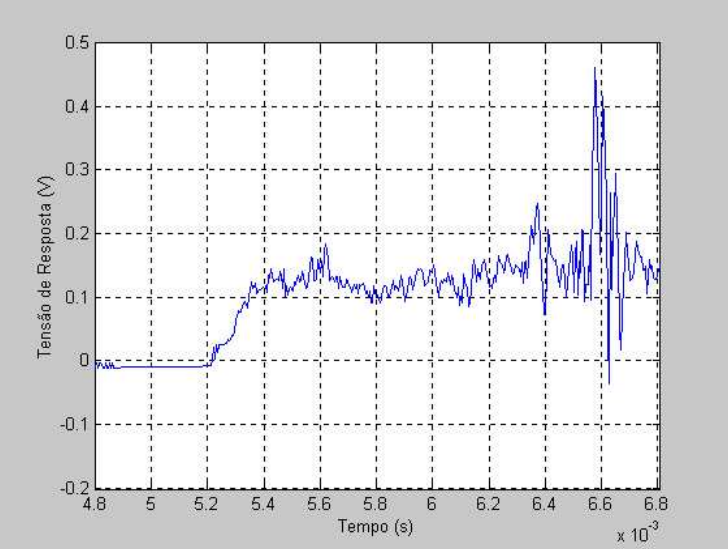


Figure 8: Typical pressure trace (voltage trace) for impact pressure.

Figures 9 and 10 show a time-lapse photograph of the luminous airflow around the mode at Mach 6.2 (high enthalpy) and 7.3 (medium enthalpy) flow conditions. At low enthalpy there is no dissociation of the air. Therefore, there is no air luminosity. Consequently no photograph was taken. It is possible to observe in figures 9 and 10 the bow shock wave structures in front of the model.

The experimental pressure ratio at the SARA static surface (Figures 9 and 10) for free stream Mach numbers of 6.2 (high enthalpy), 7.3 (medium enthalpy) and 7.8 (low enthalpy), Table 1, agrees quite well with the theoretical pressure ratio using the Modified Newtonian theory calculations, Figure 11.

For the low enthalpy the stagnation point temperature is lower than 1000 K and the experimental heat flux, obtained using the thin-film Platinum heat flux gages and developed by Toro and Leite (2002), agrees quite well with the Lees' (1956) theory, Fig. 12. For medium enthalpy the stagnation point is higher than 1000K and lower than 2000K and the air is no longer perfect gas and the heat flux close to stagnation point is higher than expected.

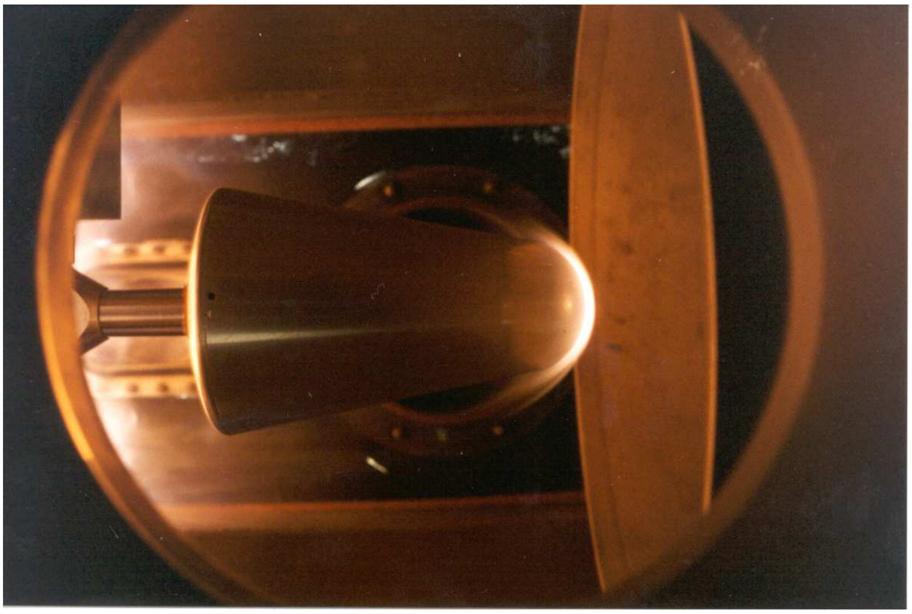


Figure 9. Open shutter photograph of the airflow luminosity upstream of a SARA model in Mach 6.2 flow

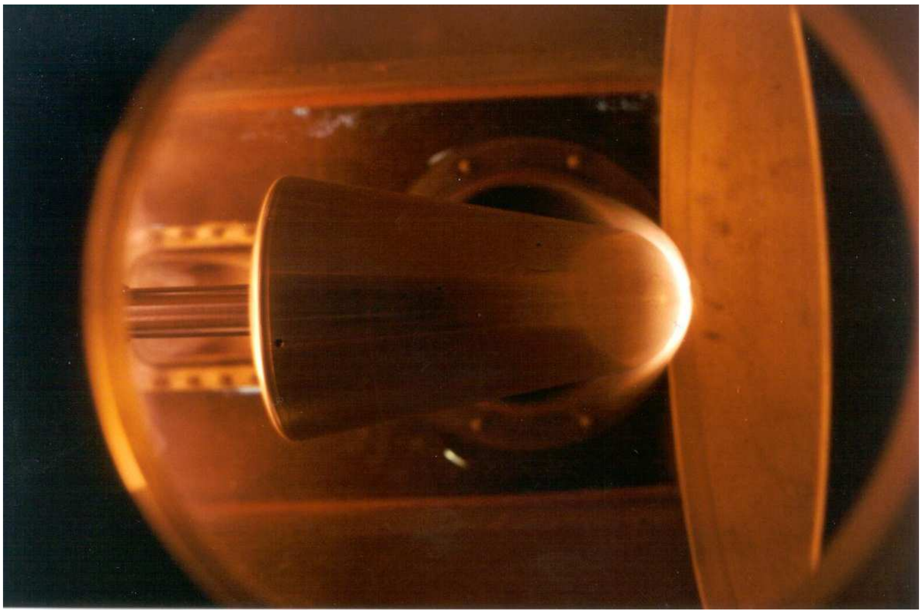


Figure 10. Open shutter photograph of the airflow luminosity upstream of a SARA model in Mach 7.3 flow

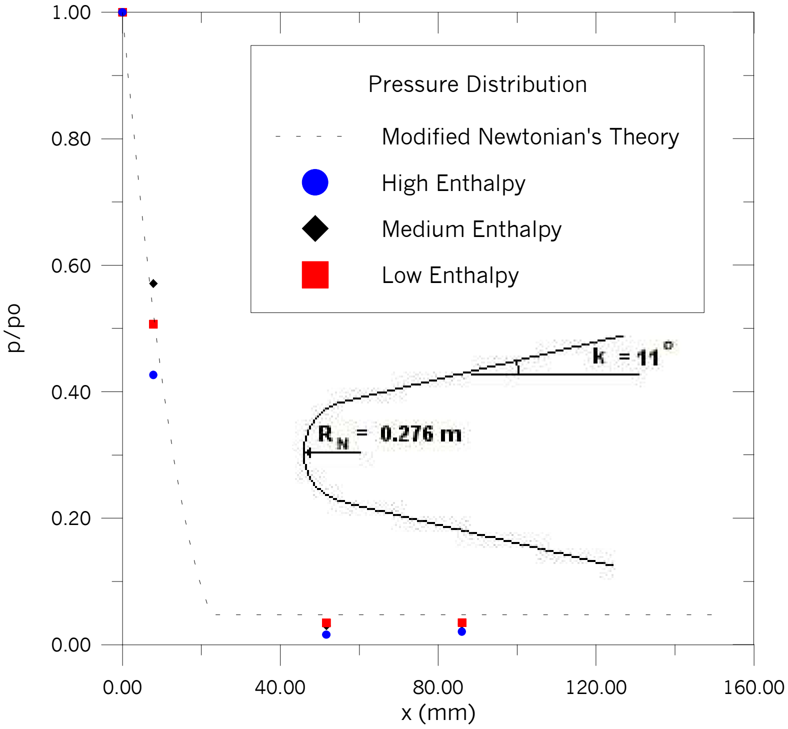


Figure 11. Pressure distribution.

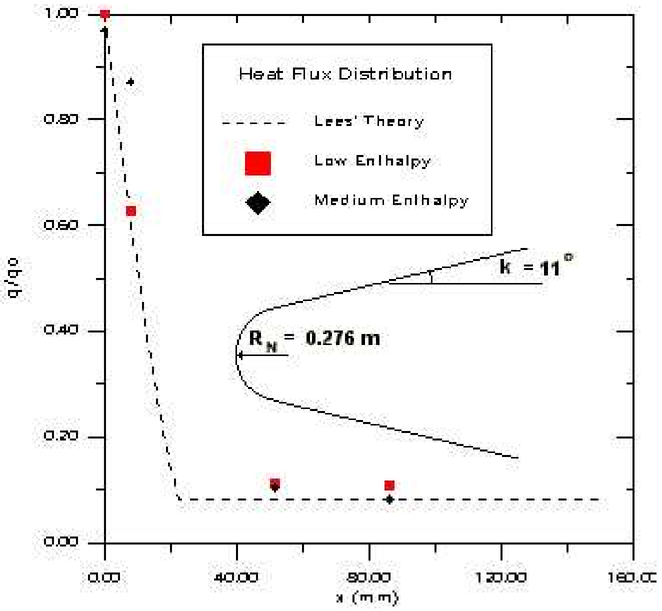


Figure 12. Heat flux distribution.

5. Conclusions

The development of experiments in a microgravity environment, using a recoverable orbital system has been proposed by Brazilian Space Agency. The system consists of a capsule shaped orbital platform. To understand the hypersonic flow characteristics of the returnable space vehicle, a 10-cm diameter Aluminum model has been fabricated and instrumented with piezoelectric pressure transducers and heat flux gages, to be investigated in the IEAv 30-cm. diameter Hypersonic Shock Tunnel at freestream Mach from 6 to 8 with stagnation temperature from 5000K to 950 K. The time-lapse photograph of the luminous airflow around the model at Mach 6.2 (high enthalpy) and 7.3 (media enthalpy) flow conditions were taken, and pressure and heat flux distributions are presented. The pressure experimental data agrees with the modified Newtonian theory and the heat flux experimental data agrees with the Lee theory.

6. Acknowledgments.

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