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EXPERIMENTAL INVESTIGATION OF TRANSONIC FLOW PAST A SOUNDING ROCKET VS-40

Pedro Geovanny Martínez Romero
Instituto Tecnológico de Aeronáutica (ITA), São José dos Campos, SP, 12228-900, Brazil
ingmartinezromero@gmail.com

Ana Cristina Avelar
Instituto de Aeronáutica de Espaço (IAE), São José dos Campos, SP, 12228-904, Brazil
anacristina.avelar@gmail.com

João Batista Falcão Filho3
Instituto de Aeronáutica de Espaço (IAE), São José dos Campos, SP, 12228-904, Brazil
falcaojbpff@iae.cta.br

Abstract. *This study is a continuation of a series of analyses carried out since 2013 on the VS-40 sounding rocket. Experimental tests were conducted in the Pilot Transonic Wind Tunnel, known as TTP, of the Institute of Aeronautics and Space (IAE), on a 1:34 scaled model of the VS-40 without back step after the frustum cone for Mach numbers ranging from 0.4 to 1.0. Reynolds number effects in the flow field over the model and consequently in the drag of the vehicle is investigate in an attempt to predict the physical phenomena occurring in a flight condition. The motivation is that the Reynolds number of flight of a real space vehicle is very high. Consequently, the laminal to turbulent boundary layer transition occurs very soon, still in the beginning of the fore-body section. On to other hand, in wind tunnel tests, the flow is expected to be mostly laminar over a great extension of the scale model. So, in order to analyze the transition strip effects in the pressure distribution over the model, tests were carried out with and without a transition strip on the model. Pressure field obtained with PSP technique will be presented for several values of Mach number as well as complex phenomena as shock waves and expansion waves will be identified from the PSP pressure field and from the Schlieren images.*

Keywords: *transonic flow, Sounding rocket, Pressure Sensitive Paint, Schlieren*

1. INTRODUCTION

Pressure measurements on model surfaces are essential in wind tunnels tests are of great importance for the understanding of the aerodynamic performance of flight vehicles as well as to provide critical information about complex flow phenomena (Liu and Sullivan, 2005). The Pressure sensitive Paint technique is a relatively new tool that has the ability to provide pressure field measurements over the entire surface of a model, and it is especially important in the transonic regime, which is part of the flight envelop of aerospace vehicles associated with several important flow features, as expansion and shock waves occurrences, flow separation and shock wave boundary layer interaction, among others.

The VS-40 is a Brazilian sounding rocket, developed in the Instituto de Aeronáutica e Espaço, IAE, stabilized aerodynamically that uses solid fuel, distributed between the first stage S40TM (4.200 kg) engine and the second stage S44M (810 kg) engine. It is a non controlled, unmanned and high performance space vehicle (Barbosa and Guimarães, 2012).

In the present study wind tests tunnel were carried out for the VS-40 model, Fig. 1. The results were obtained using several experimental methods fund in the Pilot Transonic Wind Tunnel, known as TTP, of the Institute of Aeronautics and Space (IAE) as the Pressure Sensitive Paint PSP and Schlieren. The VS-40 sounding rocket vehicle was conceived for achieving only one experimental launching for qualifying the forth stage of VLS-1 in flight conditions, in the outer space vacuum. However, the study of the rocket project showed that the vehicle was highly efficient and promising as a sounding vehicle, in terms of performance and payload volume. Consequently, it is has been used successfully with this purpose. Up to now four flights have already been carried out. The VS-40 was chosen to carry the sub-orbital SARA, a Brazilian platform for microgravity experiments, and recently, in June 2012, it was launched with the Germany experiment Sharp Edge Fight Experiment (SHEFEX II) in Andoya, Norway. Future launches expected to be realized in 2015.

Figure 1. VS-40 Sounding Vehicle.

Previous studies have already been conducted analyzing the aerodynamic sounding vehicle (Avelar, *et al*., 2014a; Avelar, *et al*., 2014b). The studies were conducted for investigating the flow pattern in the nose fairing of a sounding vehicle with a back step after the frustum. In present paper, experimental tests were carried out in the Pilot Transonic Wind Tunnel, known as TTP, of the Institute of Aeronautics and Space (IAE), on a scaled model 1:34 of the VS-40 without back step after the frustum cone for Mach numbers ranging from 0.4 to 1.0. Experimental testes with and without a transition strip on the model were carried out to analyze the transition strip effects in the pressure distribution along the centerline of the model.

Traditional measuring methods are based in pressure taps connected to remote pressure sensors to obtain pressure values for some points on the model, but this technique presents some spatial limitations when very thin wings or models with sharp corners are the object of investigation (Watkins, *et al*., 2011). The results presented in this paper were obtained with the PSP technique. The PSP technique working principle is based on the sensitivity of certain luminescent molecules, luminophores, to the presence of oxygen. A special coating is used together with digital-imaging technology to obtain global surface pressure distribution (Bell, *et al*., 2001). As represented schematically in Fig. 2, a typical PSP paint consists of sensor molecules, the luminophores, contained in an oxygen-permeable binder. When illuminated with light at an appropriated wavelength, the sensor molecules are excited to an elevated energy state. The molecules then recover to the ground state, typically, by a radiative decay (luminescence) and nonradiative decay trough the release of heat.

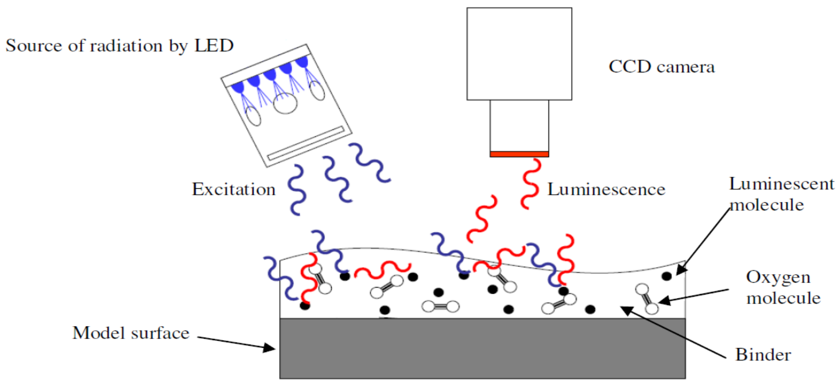


Figure 2. Physical principle of the PSP technique (extracted from ISSI website).

Some material can also return to the ground state by colliding with an oxygen molecule, a process known as "oxygen quenching". The rate of quenching is proportional to the oxygen partial pressure, which is in turn proportional to the local surface pressure. The PSP technique working principle is well documented in the literature (Engler, *et al*.,2000; Gregory, *et al.*,2008).

2. RESULTS

In this section some preliminaries results obtained up to now with the PSP techniques, pressure taps and Schlieren will be presented. Most of the PSP data are still being processed and will be presented in future works.

Firstly it will be presented some results of the pressure field obtained of the VS-40 sounding rocket model without transition strip by Mach 0.4 to Mach 1.0. The corresponding graphic shows the pressure distribution along the centerline of the model with the streamwise coordinate made non-dimensional by the rocket diameter. The color displayed on the map the results PSP shows low pressure values in magenta color and high pressure values in dark red color. In all cases the PSP pressure curve level are adjusted using the pressure taps at the model tip. As well as the pressure taps has higher precision.

Figure 3 and 4 shows the results obtained for free stream Mach number 0.4 and 0.6 respectively.

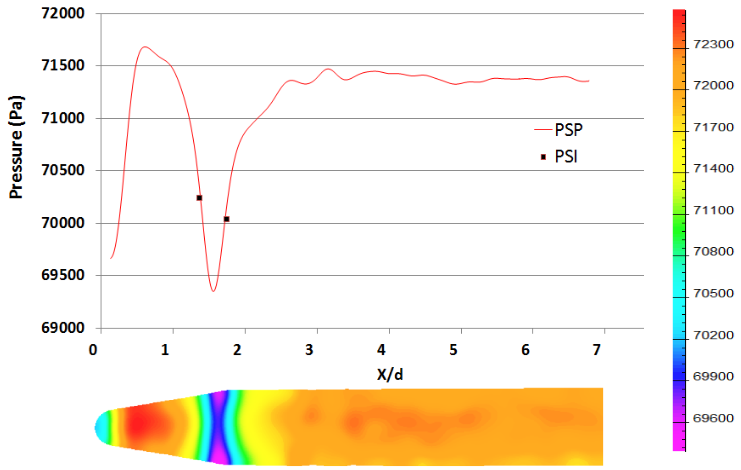


Figure 3. Pressure Sensitive Paint results for M = 0.4

The graphic shows the pressure profile on a central line over the surface of the model and the pressure taps. Fig. 3 shows an increase of the pressure in the first part of the warhead indicate on the maximum point Mach number of 0.405. At the end of the frustum cone a significant pressure drop is observed. In this case the low pressure found indicates Mach number of about 0.461.The pressure increases after the frustum cone reaching Mach number of 0.412. this value is maintained over cylindrical region.

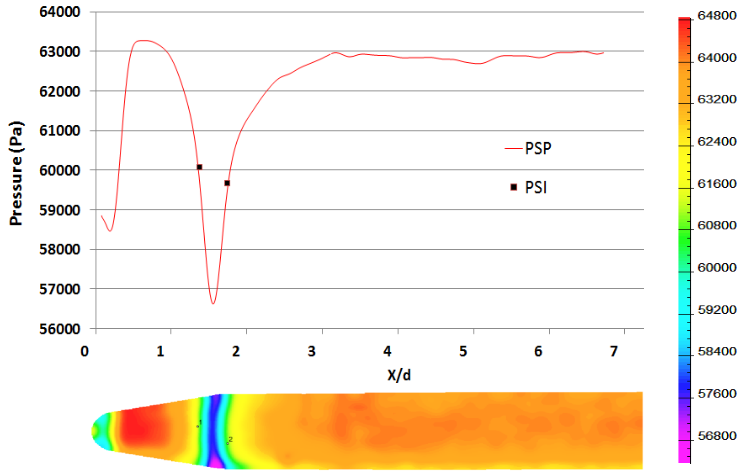


Figure 4. Pressure Sensitive Paint results for M = 0.6

The behavior of the pressure to Mach 0.6 (Fig. 4) is very similar to Mach 0.4. in this case the maximum point of the pressure the Mach number is 0.59 and the Mach on the drop the pressure is 0.72. After of the frustum cone the pressure reached Mach 0.6, and the pressure at the end of the model is equal to freestream condition. The pressure value observed resulted in Mach number of 0.598, which is practically the freestream Mach number.

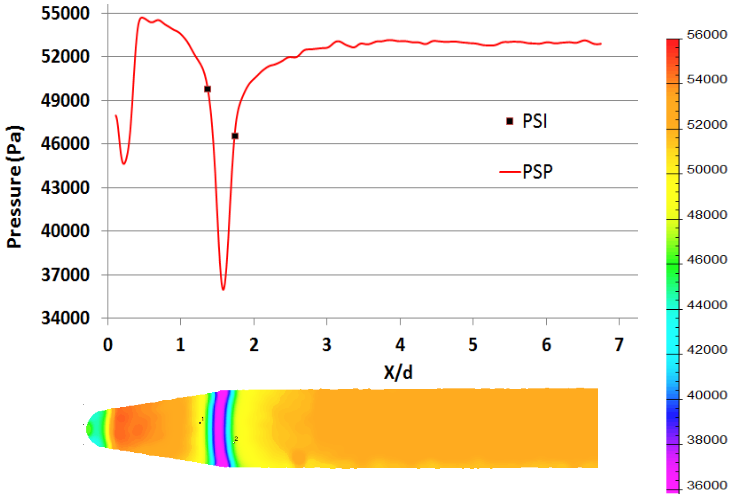


Figure 5. Pressure Sensitive Paint results for M = 0.8

Figure 5 shows the result for freestream Mach number of 0.8. After the nose the Mach number is 0.767, the frustum cone is responsible for a little pressure variation. It is interesting to observe the maximum Mach number reached was 1.2. These results indicate that the critical Mach number in the expansion region was reached for this freestream condition. After the expansion region the pressure rises to reach free stream pressure condition.

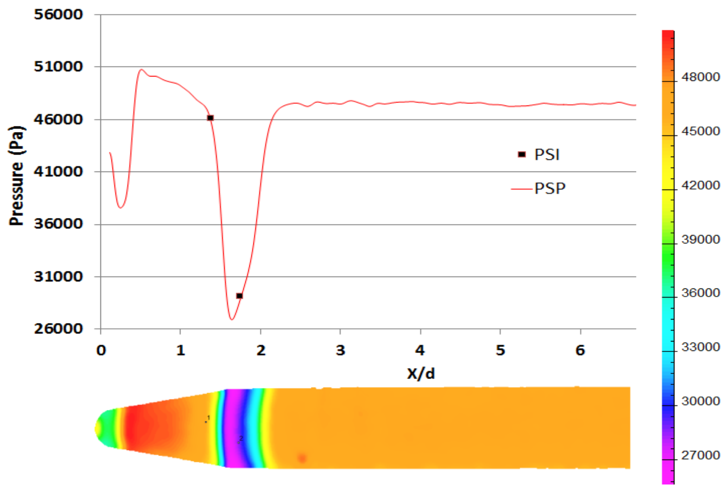


Figure 6. Pressure Sensitive Paint results for M=0.9

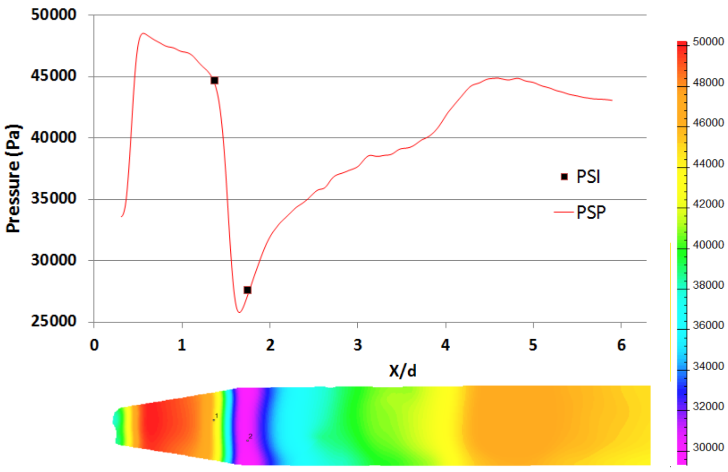
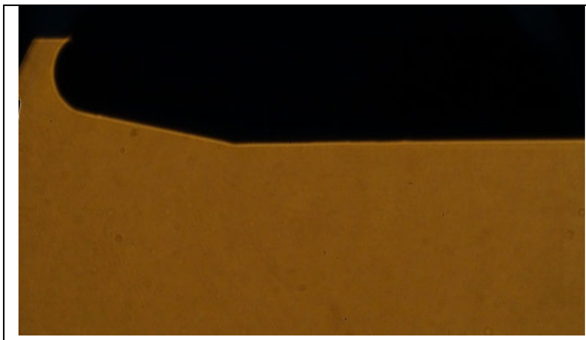
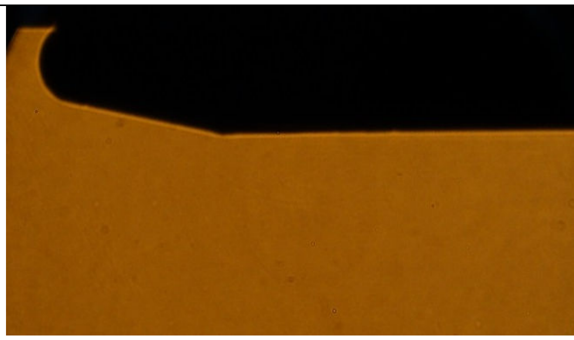


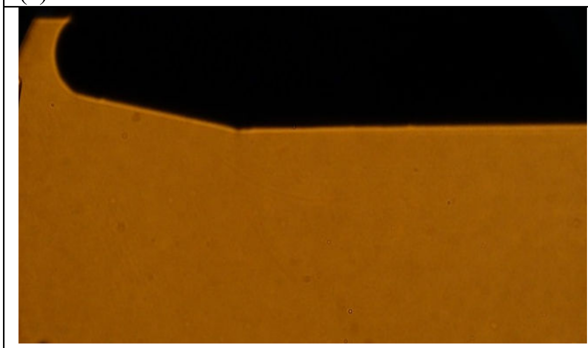
Figure 7. Pressure Sensitive Paint results for M 1.

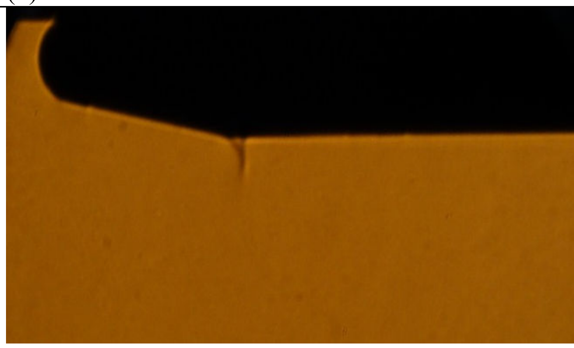
Figure 6 and 7 present the results obtained for freestream Mach number of 0.9 and 1.0, respectively. For Mach number of 0.9 the lowest pressure value at the beginning of the cylindrical section indicates Mach number value of the 1.35. It is expected a shock-wave just after this supersonic region. After the expansion region the pressure is recovered reached the freestream number. Already to freestream Mach number 1.0 can be observed a quite different result for the graphic. In this case the model has two low-pressure values. The first lowest pressure value at the beginning of the model indicates Mach number of 1.19. The second lowest pressure value just at the end of the frustum cone was Mach number of 1.38. It is followed by a complex and long shock-wave formation, reaching a final Mach number of 1.08 just followed by an expansion-wave. From that point on, the pressure rises probably because of compression waves formed to decelerate the flow reaching Mach number of 0.95.

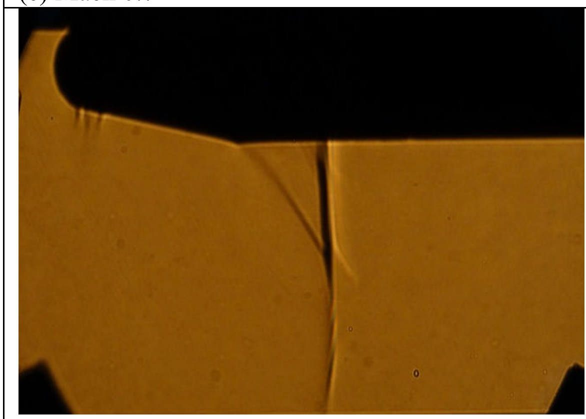
Table 1. Schlieren images for Mach number varying 0.4 to 1.0 for the model without transition strip.

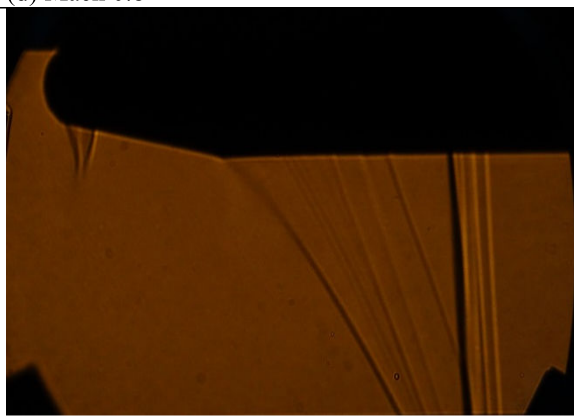

(a) Mach 0.4


(b) Mach 0.6


(c) Mach 0.7


(d) Mach 0.8


(e) Mach 0.9


(f) Mach 1.0

In order to get more insights about the complicated flow pattern, images with the Schlieren technique were carried out. Table 1 shows the images obtained for Mach number 0.4 to 1.

To freestream Mach number 0.4 and 0.6 is not observed any changes around the model. At Mach number 0.8 it is possible to observe significant expansion or shock formation, only a tiny region at the frustum cone end indicating some density gradient. As expected a shock-wave just after the frustum cone is observed for the Mach number 0.9. In the subsequent pictures, Freestream Mach number 1, one can see that the increase of the speed will provoke the expansion/shock wave group. The first formed in the region just after the vehicle noise and the second formed after of the frustum cone. Exactly the same points where images PSP presents two peaks of lower pressure values.

Now was put the transition strip about 5% of the model length obliging the transition from laminar to turbulent boundary layer, still the front surface of the vehicle. In order to analyze the influence of the transition strip over the pressure field distribution along the VS-40 sounding rocket model.

The most important results for this investigation are detailed next. Figure 8 shows the pressure distribution over the model obtained whit the technique PSP for freestream Mach number 0.6.

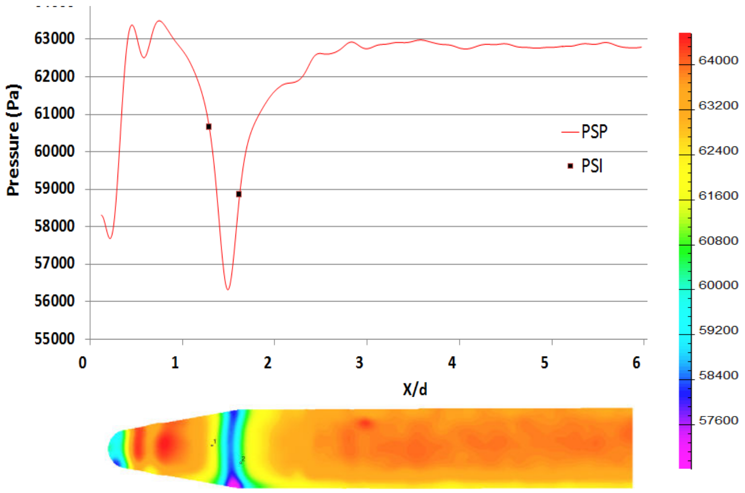


Figure 8. Pressure Sensitive Paint results for M=0.6 with transition strip.

In this case which would be expected to have a small pressure decrease along the whole conical nose region, it resulted in two distinct regions: a large pressure increase before the transition strip, followed by a pressure drop due to its presence. This can be seen in the PSP image by the red region just after the nose tip followed by the orange region.

The pressure levels in both parts before and after the pressure strip indicate Mach numbers of 0.60 and 0.57, respectively. After the transitions strip the pressure decreases reaching 56.3KPa, the Mach number 0.73, a value relatively high considering the freestream Mach number of 0.6. After the frustum cone, in a position X/d equal 3, the flow arrive the freestream pressure condition. It is interesting to observe a mild oscillation in the final region of the pressure distribution curve, which is still being investigated if it is related to the PSP technique or physical disturbance of the flow.

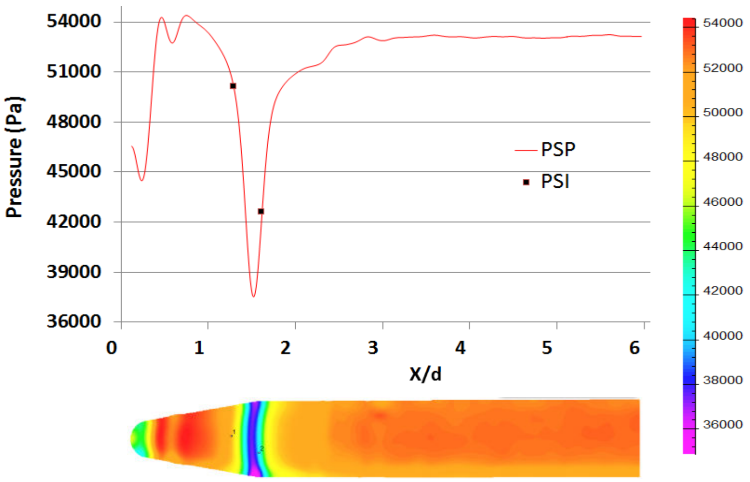


Figure 9. Pressure Sensitive Paint results for M = 0.8 with transition strip.

Figure 9 and 10 present the results obtained for freestream Mach number of 0.8 and 0.9, of model with transition strip respectively. For Mach number 0.8 Fig.9, the variation between the Mach number before and after of the transition strip is only 0.5x10-3. In this case the model reaches the critical Mach number in the expansion region, with a pressure of 37.6KPa and Mach number 1.1. Less that the same conditions without transition strip that was 34.8KPa end Mach number 1.2.

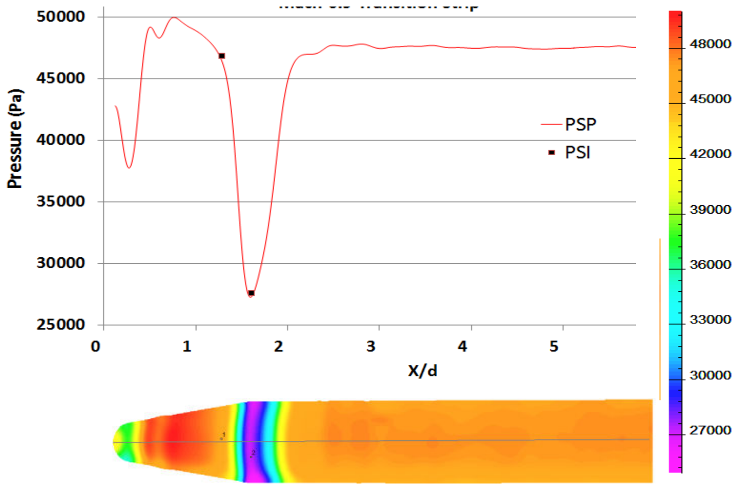


Figure 10. Pressure Sensitive Paint results for M = 0.9 with transition strip

It is observed in both the graphic and the image PSP of fig. 10 that the pressure recovery region is larger. This indicates a formation of expansion wave. Just after the nose the pressure reaches Mach number 1.1. It is expected the beginning of a expansion wave close to the model tip. In this case the lowest local pressure it is located again after the frustum cone indicates Mach number value of 1.34. From this point the pressure increases reaching the free stream condition.

The largest differences are observed in the results for freestream Mach number 1. Fig. 11. Where the pressure distribution along the model present a signicant difference from the subsonic case. Close to the model tip and after the frustum cone the flow expands. The estimated local Mach numbers at the expansion regions, just after the model tip and just after the frustum cone, are 1.21 and 1.37. Respectively. The maximum valor pressure reached was in the frustum come region for this case was 47.8KPa. Lower than the model without the transition strip with 48.2KPa. In this case after the frustum cone there are strong expansion wave formations followed by shock waves. In the PSP image it is possible to observe an enlargement of this expansion region. Because of the supersonic nature of the flow after the expansion region the pressure will no converge to freestream condition but will reach pressure of 44.7KPa, higher than the freestream pressure, corresponding to a Mach number of about 0.95. From this point on the subsonic region will accelerate until reaching freestream Mach number 1 condition.

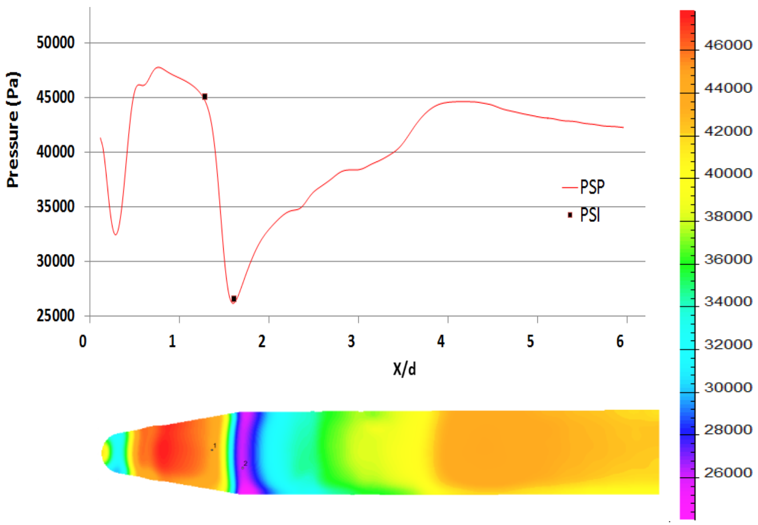
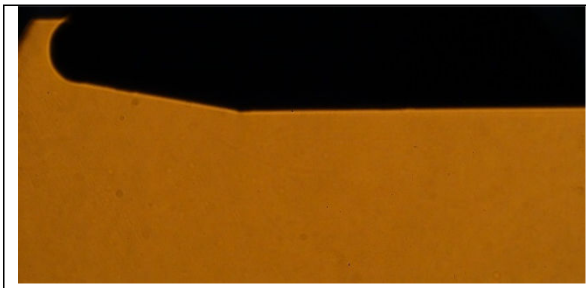
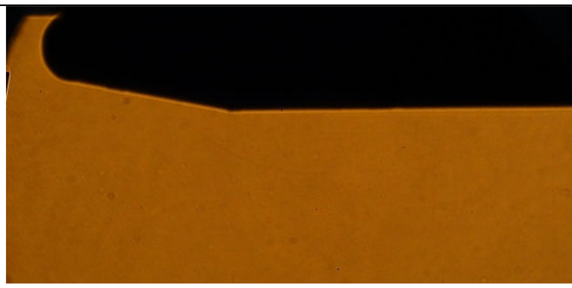


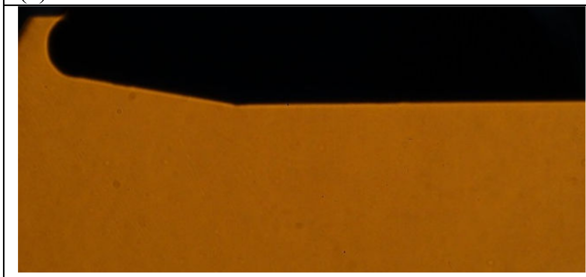
Figure 11. Pressure Sensitive Paint results with transition strip M=1.

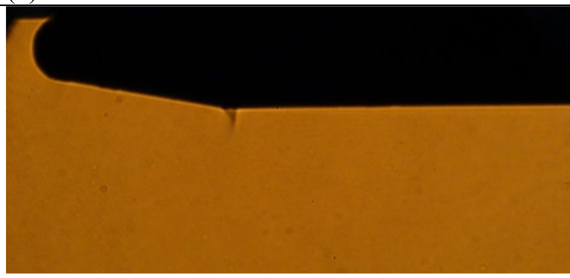
Visualization using the Schlieren technique was also conducted for model with transition strip and are presented in the table 2.

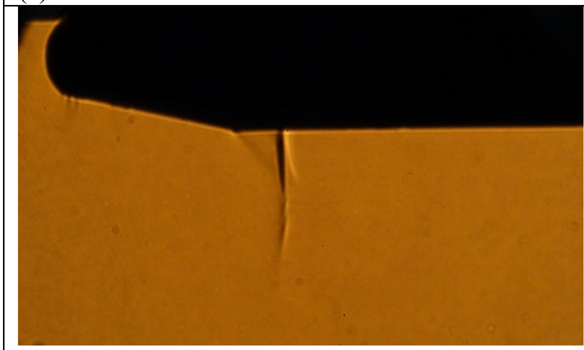
Table 2. Schlieren images of flow patterns around VS-40 fore-body section with transition strip for M=0.7 to M=1.08.


(a) Mach 0.4


(b) Mach 0.6


(c) Mach 0.7


(d) Mach 0.8


(e) Mach 0.9


(f) Mach 1.0

For free stream Mach number 0.4 and 0.6 the flow is completely subsonic and the Schlieren image do not show any shock formation, At Mach number 0.8 the image shows the beginning of the formation of a shock wave after the frustum cone. Higher density gradient are observed far Much number 0.9 and 1. For free stream Mach number 0.9 a shock-wave just after the frustum cone is observed. Another expansion/shock wave is observed after the frustum cone.

For Mach number 1 the expansion effect is stronger and a mild multiple shock wave formation is observed. In this case the shock wave has a more complex look, one can see that this effect is amplified. The image shows that the shocks are merged in one normal shock wave in a very familiar pattern of expansion followed by shock wave.

Comparing the images of Tab. 1 with the table 2, can be observed that the transition strip not change the flow behavior but decreases the intensity.

3. CONCLUSION

The procedure for an experimental analyses of the flow pattern around the VS-40 sounding vehicle without back step after the frustum cone were described, and preliminary results obtained with the three techniques used, Schlieren, PSP an pressure taps were presented. Both for the model with transition strip as for the without transition strip model the flow is completely subsonic in all regions of vehicle for free stream Mach number up to 0.7. In both cases the critical Mach number is reached for free stream Mach number of 0.8. It was observed that the transition strip not change the flow behavior along the model but decreases the intensity in the maximum and minimum static pressure point since free stream Mach number of 0.8 .

The obtained results allowed the identification of several important flow features, as expansion waves, shock waves and shocks merged, expansion wave followed by normal shock wave. In general, the PSP technique presented very well behavior in transonic regime, being very feasible for wind tunnel studies of sounding rockets in wind tunnel. The techniques Schlieren combined constitute a powerful tool for analyses of complexes phenomena.

6. ACKNOWLEDGEMENTS

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4. RESPONSIBILITY NOTICE

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